AIRPLANE MAINTENANCE MANUAL

PA-46-350P

MIRAGE

/M350

(S/N'S 4636001 AND UP)

PA-46R-350T

MATRIX

(ALL)

PIPER AIRCRAFT, INC.

PART NUMBER 761-876

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GAMA

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REVISION STATUS

1. Definitions

A. Revision

The data in the revision column is comprised of two elements:

(1) A Type of Revision Code: ORG = Original, CR = Complete Revision, and PR = Partial Revision.

NOTE: Partial Revisions (PR) are listed only until the next Complete Revision (CR) is published. Then they are removed.

(2) The Revision Date in YYMMDD format.

NOTE: The Revision Date is the date placed on each revised page. It exists to separate one version of a page from another. Revision Date does not indicate the calendar date when the revision was actually published and available to the public. However, ***this is the date that appears in the Current Revision Checklist*** in the Customer Service Information File.

B. Publication Date

Publication Date usage has varied over the years. 1937–1996 is unknown. 1997–2013 it generally was synchronized with the revision date regardless of when the revision was published. In mid 2013 and later, the Publication Date is the calendar date when the revision was actually published and available to the public.

NOTE: ***This date does not appear in the Current Revision Checklist*** in the Customer Service Information File.

2. Revisions

Revisions to this Maintenance Manual, P/N 761-876, issued July 12, 1995 are as follows:

Revision	Publication Date
ORG950712	June 19, 1996
CR981223	December 23, 1998
CR050730	July 30, 2005
CR070228	February 28, 2007
CR070828	August 28, 2007
CR090915	September 15, 2009
PR100701	July 1, 2010
PR110531	May 31, 2011
PR130115	June 1, 2013
PR140215	February 15, 2014
PR141215	April 10, 2015
PR150715	August 23, 2017
PR171130	December 19, 2017
PR190731	September 25, 2019
PR200221	May 28, 2020
PR200625	June 25, 2020
PR210315	May 3, 2021

3. Availability

This maintenance manual, related inspection reports and manuals, service publications (SB, SL, etc.) and other Piper publications are available as described in the Owner Publications Catalog (part of the Customer Service Information File, see below).

Consult the "Customer Service Information File" (a free download from the Piper Aircraft, Inc. website at http://www.piper.com/technical-publications-documents/) to verify that you have the latest revision.

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INTRODUCTION

LIST OF EFFECTIVE PAGES

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INTRODUCTION

TABLE OF CONTENTS

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INTRODUCTION

1. Instructions for Continued Airworthiness

WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

The PIPER PA-46-350P Malibu Mirage / M350 and PA-46R-350T Malibu Matrix Maintenance Manual constitutes the Instructions for Continued Airworthiness as required by Federal Aviation Regulations (FAR) Part 23, Appendix G. Chapter 4 contains the Airworthiness Limitations section (4-00-00) and the Inspection Program is in Chapter 5 (5-20-00).

2. General

This publication is prepared in accordance with the General Aviation Manufacturers Association (GAMA) Specification No. 2, with respect to the arrangement and content of the System/Chapters within the designated Chapter/Section-numbering system.

WARNING: USE ONLY GENUINE PIPER AIRCRAFT PARTS OR PIPER AIRCRAFT APPROVED PARTS OBTAINED FROM PIPER APPROVED SOURCES, IN CONNECTION WITH THE MAINTENANCE AND REPAIR OF PIPER AIRPLANES.

This manual generally does not contain hardware callouts for installation. Hardware callouts are only indicated where a special application is required. Even then, confirm the correct hardware per the PA-46-350P Parts Catalog, P/N 761-878 or the PA-46-350T Parts Catalog, P/N 766-885, and FAR 43 for proper utilization.

Genuine PIPER parts are produced and inspected under rigorous procedures to ensure airworthiness and suitability for use in PIPER airplane applications. Parts purchased from sources other than PIPER, even though identical in appearance, may not have had the required tests and inspections performed, may be different in fabrication techniques and materials, and may be dangerous when installed in an airplane.

Reworked, salvaged or those parts obtained from non-PIPER approved sources, which the service history is unknown or cannot be authenticated, may have been subjected to unacceptable stresses or temperatures or have other hidden damage, not discernible through routine visual or usual nondestructive testing. This may render the part, component or structural assembly, even though originally manufactured by PIPER AIRCRAFT unsuitable and unsafe for airplane use.

PIPER AIRCRAFT, INC. expressly disclaims any responsibility for malfunctions, failures, damage or injury caused by use of non-PIPER approved parts.

NOTE: PIPER AIRCRAFT, INC. expressly reserves the right to supersede, cancel and/or declare obsolete any part, part numbers, kits or publication that may be referenced in this manual without prior notice.

Also, Piper Aircraft, Inc. may possess manufacturer's data which defines minimum type design beyond what may be assumed by an authorized repair entity. When a repair is proposed, it is the responsibility of the repairer per AC 43.13-1 to determine that the proposed repair is not contrary to manufacturer's data. The repairer or aircraft owner or his agent should contact Piper directly to determine that a proposed repair is not in conflict with minimum type design capability.

If you have any question concerning the care of your airplane, be sure to include the airplane serial number in any correspondence to Piper.

3. Effectivity

This maintenance manual is effective for PA-46-350P airplane serial numbers 4636001 and up, and PA-46R-350T airplane serial numbers 4692001 and up. This encompasses the following model years:

NOTE: The following is provided as a general reference only.

PA-46-350P Mirage

Model Year	Serial Numbers
1995	4636001 thru 4636020
1996	4636021 thru 4636076
1997	4636077 thru 4636131
1998	4636132 thru 4636186
1999	4636187 thru 4636247
2000	4636248 thru 4636313
2002	4636299, and 4636314 thru 4636338
2003	4636339 thru 4636348
2004	4636349 thru 4636363
2005	4636364 thru 4636374
2006	4636375 thru 4636399
2007	4636400 thru 4636433
2008	4636434 thru 4636451
2009	4636452 thru 4636459, and 4636461
2010	4636460, and 4636462 thru 4636488
2011	4636489 thru 4636502
2012	4636503 thru 4636559
2013	4636560 thru 4636600
2014	4636601 thru 4636632 4636634 thru 4636637
2015	4636638 thru 4636651

PA-46-350P M350

Model Year	Serial Numbers
2015	4636652 thru 4636673
2016	4636633, and 4636674 thru 4636710
2017	4636711 thru 4636719
2018	4636720 thru 4636739
2019	4636740 thru 4636760
2020	4636761 thru 4636775
2021	4636776 and up

PA-46R-350T Matrix

Model Year	Serial Numbers
2008	4692001 thru 4692086
2009	4692087 thru 4692133
2010	4692134 thru 4692155
2011	4692156 thru 4692165
2012	4692166 thru 4692183
2013	4692184 thru 4692197
2014	4692198 thru 4692210
2015	4692211 thru 4692213
2016	4692214

4. Serial Number Explanation

Example:

46 36 001

TYPE CERTIFICATE DESIGNATION

SEQUENCE NUMBER

MODEL CODE
36 = PA-46-350P MALIBU MIRAGE / M350
92 = PA-46R-350T MALIBU MATRIX

5. Assignment of Subject Material

This publication is divided into industry standard, three element, numeric subject groupings as follows:

A. System/Chapter - The various groups are broken down into major systems such as Environmental Systems, Electrical Power, Landing Gear, etc. They are assigned a number, which becomes the first element of the standardized numbering system. Thus, the element “28” of the number 28-40-01 refers to the chapter “Fuel”. Everything concerning the fuel system will be covered in this chapter.

B. Sub-System/Section - The major systems/chapters of an airplane are broken down into subsystems. These sub-systems are identified by the second element of the standard numbering system. The element “40” of the number 28-40-01 concerns itself with the indicating section of the fuel system.

C. Unit/Subject - The individual units within a sub-system/section may be identified by the third element of the standard numbering system. The element “01” of the number 28-40-01 is a subject designator. This element is assigned at the option of the manufacturer and is normally zeroed out by PIPER.

Refer to paragraph 14, Chapter/Section Index Guide, for a complete breakdown and list. The material is arranged in ascending numerical sequence.

6. Pagination

The Chapter - Section (i.e. - 28-40-00) numbering system (explained above) forms the primary page numbering system for this manual. Within each Section, pages are numbered consecutively beginning with Page 1 (i.e. - 28-40-00, Page 1).

7. Aerofiche Grid Numbering

Piper has ceased production of all Aerofiche (i.e., microfiche) products. The Aerofiche grid numbers have been removed.

8. Identifying Revised Material

A vertical line (i. e. - change bar) along the left-hand margin of the page (or text column) is used to identify revised text or illustrations.

Example.

A change bar in the left-hand margin opposite the footer (i.e. - chapter/section/subject, page number and date), indicates that the text was unchanged but the material was relocated to a different page.

Change bars in the individual chapter Tables of Contents do not indicate a change to that page, but rather that the information in the actual paragraph has changed.

NOTE: Change bars are not used in the title pages, list of effective pages, index, or Wiring Diagram Cross-Reference.

A. 2009–2011

A revision to a page is defined as any change to the printed matter that existed previously. Revisions, additions and deletions are identified by a vertical line (i. e. - change bar) along the left-hand margin of the page opposite only that portion of the printed matter that was changed.

B. 2013 and later

A revision to a page is defined as a change to the text or illustrations that existed previously. Revisions, additions and deletions are identified by a vertical line (i. e. - change bar) along the left-hand margin of the page opposite only the text or illustration that was changed. Reformatted, but otherwise unchanged, text is not identified by a change bar.

9. Indexing

An alphabetically arranged Subject Index follows this introduction to assist the user in locating desired information. In addition, each System/Chapter begins with an individual Table of Contents.

10. List of Effective Pages

Each System/Chapter has a List of Effective Pages preceding the Table of Contents to identify the effective revision date for each page in that chapter.

11. Warnings, Cautions and Notes

These adjuncts to the text are used to highlight or emphasize important points when necessary. WARNINGS call attention to use of materials, processes, methods, procedures or limits which must be followed precisely to avoid injury or death to persons. CAUTIONS call attention to methods and procedures which must be followed to avoid damage to equipment. NOTES call attention to methods which make the job easier. Warnings and Cautions precede and Notes follow the text to which they apply.

12. Accident/Incident Reporting

To improve our Service and Reliability system and aid in Piper's compliance with FAR 21.3, knowledge of all incidents and/or accidents must be reported to Piper immediately. To expedite and assist in reporting all incidents and accidents, Piper Form 420-01 has been created. See latest revision of Service Letter 1041 for latest version of this form. This procedure is to be used by all Dealers, Service Centers and Repair Facilities.

13. Supplementary Publications

The following is a list of publications providing servicing, overhaul and parts information on various components on these airplanes, which you should use to supplement this manual.

A. Piper Publications

	Part Number
(1) Parts Catalogs	
(a) PA-46-350P Malibu Mirage	
1) Meggitt / Avidyne S/N's 4636001–4636463, less 4636460	761-878
2) Garmin G1000 S/N's 4636460, 4636463 & up	766-884
(b) PA-46R-350T Malibu Matrix	
1) Avidyne S/N's 4692001–4692133, 4692141, 4692149, & 4692153	767-069
2) Garmin G1000 S/N's 4692134 & up, less 4692141, 4692149, & 4692153	766-885
(2) Periodic Inspection Report Form	767-011
(3) Progressive Inspection Manual (50 hour)	767-013
(4) Wiring Diagram Manual	767-116

B. Vendor Publications

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY.

(1) AIR CONDITIONING COMPRESSOR:

Vendor:	Sanden International (USA), Inc. 601 South Sanden Blvd. Wylie, Texas 75098 http://www.sanden.com/	PH: FAX:	(972) 442-8400 (972) 442-8700

(2) AIR CONDITIONING EVAPORATORS AND BLOWERS

Vendor:	Enviro Systems, Inc. P.O. Box 1404 Seminole, Oklahoma 74868	PH:	(405) 382-0731

(3) ALTERNATOR

Vendor:	Hartzell Engine Technologies 2900 Selma Hwy Montgomery, Alabama 36108 http://www.hartzell.aero/brands/	PH: FAX:	(877) 359-5355 (334) 386-5410

(4) AUTOFLIGHT

Vendor(s):	Honeywell One Technology Center 23500 W. 105th St., M/D #45 Olathe, Kansas 66061-1950 http://www.bendixking.com/	(or)	Genesys Aerosystems One S-TEC Way Mineral Wells, TX 76067 PH: (817) 215-7600 http://genesys-aerosystems.com/

or,

See Garmin under “Integrated Avionics Systems (IAS)” on page INTR8

(5) BATTERIES

(a) Aircraft Main

Vendor:	Gill/Teledyne Battery Products 840 W. Brockton Ave. Redlands, California 92374 http://www.gillbatteries.com/	PH:	(800) 456-0070

or,

Vendor:	Concorde Battery Corp. 2009 San Bernardino Road West Covina, CA 91790 http://www.concordebattery.com	PH:	(626) 813-1234

RG® Series Main Aircraft Battery CMM: Document No. 5-0171

(b) Standby Attitude Indicator Emergency Power Supply (S/N's 4636375 and Up only)

Vendor:	Avionics Systems L-3 Communications 5353 52nd Street, SE Grand Rapids, MI 49512 http://www.as.l-3com.com email: avionics.techpubs@l-3com.com	PH:	(800) 453-0288 (616) 949-6600
Service Letter:	SL-120F, or latest revision, PS-834 Battery Charging and Capacity Testing.		

(6) BRAKES

Vendor:	Parker Hannifin Corp. Aircraft Wheel and Brake Division 1160 Center Road Avon, Ohio 44011 http://www.parker.com/	PH:	(800) 272-5464

(7) DEICE SYSTEM (PNEUMATIC)

Vendor:	De-Icing and Specialty Systems Goodrich Corporation 1555 Corporate Woods Parkway Uniontown, Ohio 44685-8799	PH: FAX:	(330) 374-3040 (330) 374-2290
Technical Assistance:	Email: dssd.support@goodrich.com http://www.goodrichdeicing.com/	PH: FAX:	(800) 334-2377 (330) 374-3743 (330) 374-2290
Black Standard Pneumatic De-Icer Installation, Maintenance and Repair Manual:	ATA 30-10-31		
SmartBoot System:	ATA 30-10-65		

(8) ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

Vendor:	Meggitt Avionics, Inc. 10 Ammon Drive Manchester, NH 03103-7406 http://www.meggitt.com/	PH: FAX:	(603) 669-0940 (603) 669-0931

(9) EMERGENCY LOCATOR TRANSMITTER

Vendor:	Artex Aircraft Supplies 14405 Keil Road NE Aurora, Oregon 97002 http://www.acrelectronics.com/products/artex/	PH:	(800) 547-8901

(10) ENGINE

Vendor:	Lycoming Engines 652 Oliver Street Williamsport, Pennsylvania 17701 http://www.lycoming.com/	PH: FAX:	(717) 323-6181 (717) 327-7101
Overhaul Manual:	Direct Drive Models - P/N 60294-7		
Parts Catalog:	TIO-540-AE2A - P/N PC-315-7		
Operator's Manual:	TIO-540-AE2A Engines - P/N 60297-27		

(11) FIRE EXTINGUISHER (PORTABLE)

Vendor:	H3R Inc. 43 Magnolia Ave. #4 San Francisco, California 94123-2911 http://www.h3r.com/index.htm	PH:	(800) 249-4289

(12) FORCE APPLICATOR KIT

Vendor:	Safe Flight Instrument Corp. 20 New King Street White Plains, New York 10602	PH:	(914) 946-9500

(13) FUEL BOOST PUMP

Vendor:	Parker Hannifin Corp. Nichols Airborne Division 711 Taylor Street Elyria, Ohio 44036 http://www.parker.com	PH:	(800) 272-7537

(14) GROUND BLOWER (OPTIONAL)

Vendor:	Electro-Mech Inc. 2600 S. Custer Wichita, Kansas		

(15) GEAR LOCKING ACTUATORS, NOSE, GEAR DOOR ACTUATOR, HYDRAULIC PUMP, AND ALL HYDRAULIC COMPONENTS

Vendor:	Parker Hannifin Corp. (See "Brakes" on page INTR6)		
or,			
	Triumph Group (Frisby Aerospace) 4520 Hampton Road, Clemmons, NC 27012 http://www.triumphgroup.com/	 PH: FAX:	 (336) 766-9036 (336) 766-9040

(16) HI-LOK FASTENERS AND TOOLS

Vendor:	Hi-Shear Corporation 2600 Skypark Drive Torrance, California 90509	PH:	(213) 326-8110

(17) INTEGRATED AVIONICS SYSTEMS (IAS)

Vendor:	Garmin International 1200 East 151ST Street Olathe, KS 66062 http://www.garmin.com	PH:	(913) 397-8200

(a) S/N's 4636460, 4636463 thru 4636632, 4636634–4636651 and 4692134 and up:

G1000 Cockpit Reference Guide for Piper PA-46-350P MIRAGE/ PA-46R-350T MATRIX: 190-01107-00

G1000 Pilot's Guide for Piper PA-46-350P MIRAGE/ PA-46R-350T MATRIX: 190-00763-0X

(b) S/N's 4636633, 4636652–4636715, 4636717–4636719

G1000 Cockpit Reference Guide for Piper PA-46 Mirage: 190-01888-00

G1000 Pilots Guide for Piper PA-46 Mirage: 190-01889-00

(c) S/N's 4636716, 4636720 and up

G1000 NXi Cockpit Reference Guide for Piper M350: 190-02361-00

G1000 NXi Pilots Guide for Piper M350: 190-02362-00

(18) INTEGRATED FLIGHT DISPLAY SYSTEMS (IFDS)

Vendor:	Avidyne Corporation 55 Old Bedford Road Lincoln, MA 01773 http://www.avidyne.com/	PH:	(800) 284-3963

(19) LIGHTS - NAVIGATION/STROBE, STANDBY/MAP

Vendor:	Whelen Engineering Co. Inc. Route 145, Winthrop Rd. Chester, Connecticut 06412 http://www.whelen.com/	PH: FAX:	(860) 526-9504 (860) 526-2009

(20) MAGNETOS

Vendor:	Slick Aircraft Products Champion Aerospace P.O. Box 686 1230 Old Norris Road Liberty, SC 29657 http://www.championaerospace.com/	PH:	(904) 772-1909
Installation, Operation and Maintenance Instructions:	F1100 MASTER SERVICE MANUAL, 4300/6300 SERIES MAGNETO MAINTENANCE AND OVERHAUL MANUAL - L-1363		

(21) NAVIGATION, COMMUNICATIONS, AND GPS (NAV/COM/GPS)

See Garmin under "Integrated Avionics Systems (IAS)" on page INTR8

(22) OXYGEN SYSTEM

Vendor:	AVOX Systems (Scott Aviation) 225 Erie Street Lancaster, NY 14086 http://www.zodiacaerospace.com/en/zodiac-oxygen-systems-us	PH:	(716) 683-5100
or,	AEROX Aviation Oxygen Systems 206 Ossipee Trail, P.O. Box 533 Limington, ME 04049 http://www.aerox.com/	PH:	(800) 237-6902
Component Maintenance Manual (CMM) for 4110-121 Series:	Manual No. 35-00-02		

(23) PRESSURIZATION VALVES; SAFETY AND OUTFLOW VALVE & CONTROLLER VALVE

Vendor:	AlliedSignal Aerospace Aerospace Equipment Systems 2525 W. 190th Street Torrance, CA 90504 https://aerospace.honeywell.com/	PH: FAX:	(310) 323-9500 (310) 512-2221

(24) PROPELLER DEICE

Vendor:	Goodrich Corporation See “Deice System (Pneumatic)” on page INTR6.
or,	
	Hartzell Propeller Inc. See “Propeller And Propeller Governor” on page INTR9.

(25) PROPELLER AND PROPELLER GOVERNOR

Vendor:	Hartzell Propeller Inc. One Propeller Place Piqua, OH 45356-2634 http://hartzellprop.com/	PH: FAX:	(937) 778-4200 (937) 778-4391
Propeller Overhaul and Maintenance Manual:	Manual No. 113		
Propeller Owner’s Manual:	Manual No. 115 or No. 145		
Aluminum Blade Overhaul Manual:	Manual No. 133		
Composite Blade Maintenance Manual:	Manual No. 135		
Propeller Ice Protection System Manual:	Manual No. 180		
Propeller Ice Protection System Component Maintenance Manual:	Manual No. 181		
Propeller Electrical De-ice Boot Removal and Installation Manual:	Manual No. 182		

(26) RADAR (WEATHER), RDR 2000

Vendor:	Honeywell (Bendix/King) (See “Autoflight” on page INTR5)		
Maintenance Manual:	Bendix/King ART 2000 P/N 006-05332-0001		

(27) RADIO COOLING BLOWER

(See Enviro Systems under “Air Conditioning Evaporators And Blowers” on page INTR5)

(28) STANDBY ATTITUDE INDICATOR

Vendor:	Mid-Continent Instruments Co., Inc. 9400 E. 34th Street N. Wichita, KS 67226 http://www.mcico.com/	PH: FAX:	(800) 821-1212 (316) 630-0101 (316) 630-0723
Installation Manual and Operating Instructions:	Manual No. 9015762		

(29) STARTER

Vendor:	Hartzell Engine Technologies (See “Alternator” on page INTR5)		
or,			
	Sky-Tec Partners, Ltd. 350 Howard Clemmons Rd. Granbury, TX 76048 http://www.skytecair.com	PH: FAX:	(800) 476-7896 (817) 573-2252

(30) TURBOCHARGER

Vendor:	Lycoming Engines (See “Engine” on page INTR7)		

(31) VACUUM PUMP

Vendor:	Aero Accessories, Inc. 1240 Springwood Avenue Gibsonville, NC 27249 http://www.aeroaccessories.com/index.html	PH:	(800) 822-3200

(32) VACUUM REGULATORS

Vendor:	Parker Hannifin Corp. (see “Fuel Boost Pump” on page INTR7)		

14. Chapter/Section Index Guide

NOTE: The following GAMA Specification No. 2 standard chapters are not included in this Maintenance Manual: 36, 38, 49, 53, 54, 60, 72, 75, 78, and 83. These chapters are omitted because the subject system is either: not installed in these airplanes; adequately covered in vendor or other manuals; or, for ease of use, has been combined with another chapter.

CHAPTER	SECTION	TITLE
4	AIRWORTHINESS LIMITATIONS	
	00	Airworthiness Limitations
5	TIME LIMITS/MAINTENANCE CHECKS	
	00	General
	10	Time Limits
	20	Scheduled Maintenance Checks
	30	Special Inspections
	50	Unscheduled Maintenance Checks
6	DIMENSIONS AND AREAS	
	00	General
	10	Principal Dimensions
	20	Reference Lines
	40	Access Provisions
7	LIFTING AND SHORING	
	10	Jacking
8	LEVELING AND WEIGHING	
	10	Weighing and Balancing
	20	Leveling
9	TOWING AND TAXIING	
	00	General
	10	Towing
	20	Taxiing
10	PARKING AND MOORING	
	10	Parking
	20	Mooring
11	PLACARDS AND MARKINGS	
	00	General
	20	Exterior Placards and Markings
	30	Interior Placards and Markings

INDEX

Index page number references are coded.

The first four digits of the page number code are the Chapter and Section and the remaining digits are the consecutive page number within that Section - e.g., 28-10-00, page 23 would appear on these pages as “281023”.

A

B

C

D

E

H

I

J

L

M

O

P

R

S

T

V

W

CHAPTER

4

AIRWORTHINESS LIMITATIONS

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CHAPTER 4

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
4-List of	1	Jul 15/15			
Effective Pages	2	Sep 15/09			
4-Table of Contents	1	Jul 15/15			
	2	Sep 15/09			
4-00-00	1	Jul 15/15			
	2	Jan 15/13			

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CHAPTER 4 - AIRWORTHINESS LIMITATIONS

TABLE OF CONTENTS

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AIRWORTHINESS LIMITATIONS

NOTE: The Airworthiness Limitations section is FAA approved and specifies maintenance required under §§ 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA-approved.

1. Approved Mandatory Replacement Times for Type Certification

(PIR-TCDS A25SO, Rev. 25.)

The following limitations, related to fatigue life of the airplane and its components, have been established for the PA-46-350P / PA-46R-350T.

A. Refer to “Chart 1” for Structural Life Limits.

B. See “Chart 2” for Component Life Limits.

C. The safe life limit of propeller blades is unlimited.

2. Approved Mandatory Structural Inspection Intervals

None.

3. Inspection Procedures for those Approved Mandatory Structural Inspection Items

None.

CHART 1
STRUCTURAL LIFE LIMITS

Model	Structural Component	Life Limit (Flight Hours)
PA-46-350P	Pressurized Structure • Fuselage Assembly - P/N 89600-002, -003, -005, -007, -009, -014, -015	10,145
	Wing, Wing Carry-Through, and their Attaching Structure • Wing Assembly - P/N 89640-005, -006, -011 or P/N 106896-002 • Main Spar Angle - Upper Attach - P/N 83007 (Fwd), P/N 83062 (Aft) • Main Spar Angle - Lower Attach - P/N 83008 (Fwd), P/N 83063 (Aft) • Fuselage Frame - Fitting - Forward Spar - Attach - P/N 82455 • Wing - Fitting - Forward Spar - Attach - P/N 83019 • Fuselage Frame - Fitting - Aft Spar - Attach - P/N 82471 • Wing - Fitting - Aft Spar - Attach - P/N 83024	15,580
PA-46R-350T	Wing, Wing Carry-Through, and their Attaching Structure • Wing Assembly - P/N 89640-010, -012 or P/N 106896-002 • Main Spar Angle - Upper Attach - P/N 83007 (Fwd), P/N 83062 (Aft) • Main Spar Angle - Lower Attach - P/N 83008 (Fwd), P/N 83063 (Aft) • Fuselage Frame - Fitting - Forward Spar - Attach - P/N 82455 • Wing - Fitting - Forward Spar - Attach - P/N 83019 • Fuselage Frame - Fitting - Aft Spar - Attach - P/N 82471 • Wing - Fitting - Aft Spar - Attach - P/N 83024	15,580
	Horizontal Stabilizer • Horizontal Stabilizer Assembly - P/N 83404-002	15,580
	Vertical Fin • Vertical Fin Assembly - P/N 83401-009, -011	15,580

CHART 2
COMPONENT LIFE LIMITS

Model	Component	Life Limit (Flight Hours)
PA-46-350P	T.I.T. Probe (P/N 481-389, 481-392, or 686-216) *	250
PA-46R-350T	T.I.T. Probe (P/N 686-216) *	250

* see AD2011-06-10.

CHAPTER

5

TIME LIMITS / MAINTENANCE CHECKS

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CHAPTER 5

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
5-List of Effective Pages	1	Mar 25/21	5-30-00	1	Mar 25/21
	2	Sep 15/09		2	Mar 25/21
				3	Mar 25/21
5-Table of Contents	1	Mar 25/21		4	Mar 25/21
	2	Nov 30/17		5	Mar 25/21
				6	Mar 25/21
5-00-00	1	Feb 21/20		7	Mar 25/21
	2	Sep 15/09		8	Mar 25/21
				9	Mar 25/21
5-10-00	1	Jan 15/13		10	Mar 25/21
	2	Sep 15/09		11	Mar 25/21
				12	Mar 25/21
5-20-00	1	Feb 21/20			
	2	Sep 15/09	5-50-00	1	Mar 25/21
	3	Feb 21/20		2	Dec 15/14
	4	Mar 25/21		3	Nov 30/17
	5	Mar 25/21		4	Nov 30/17
	6	Mar 25/21		5	Mar 25/21
	7	Mar 25/21		6	Mar 25/21
	8	Mar 25/21		7	Mar 25/21
	9	Mar 25/21		8	Mar 25/21
	10	Mar 25/21			
	11	Mar 25/21			
	12	Mar 25/21			
	13	Mar 25/21			
	14	Mar 25/21			
	15	Mar 25/21			
	16	Mar 25/21			
	17	Mar 25/21			
	18	Mar 25/21			
	19	Mar 25/21			
	20	Mar 25/21			
	21	Mar 25/21			
	22	Mar 25/21			
	23	Mar 25/21			
	24	Mar 25/21			
	25	Mar 25/21			
	26	Mar 25/21			

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CHAPTER 5 - TIME LIMITS / MAINTENANCE CHECKS

TABLE OF CONTENTS

CHAPTER 5 - TIME LIMITS / MAINTENANCE CHECKS

TABLE OF CONTENTS (continued)

GENERAL

Piper Aircraft, Inc. (Piper) takes a continuing interest in having the owner get the most efficient use from his airplane, and keeping the airplane in the best mechanical condition. To that end, Piper publishes a recurring maintenance schedule which is supplemented with Service Bulletins, Service Letters and Service Spares Letters as required.

A. The recurring maintenance schedule for the PA-46-350P Malibu Mirage/M350 and PA-46R-350T Malibu Matrix is provided in 5-20-00.

B. Piper Service Bulletins are of special importance and Piper considers compliance mandatory.

C. Service Letters deal with product improvements and service hints pertaining to the affected aircraft. Owners should give careful attention to service letter information so they can ensure their airplane is properly serviced and kept up to date with the latest changes.

D. Service Spares Letters offer improved parts, kits and optional equipment which were not originally available. These may be of interest to the owner.

E. Service Bulletins, Service Letters and Service Spares Letters are emailed to Piper Dealers/Service Centers. Owners are encouraged to download these service publications from Piper's website. See Revision Status on page iii for the current web address.

NOTE: Piper mails flight manual (AFM / POH) revisions to the registered owner's name and address as shown on the Aircraft Registration Certificate. If the aircraft is based and/or operated at a different location (or locations) and/or by a person (or persons) other than those recorded on the aircraft registration, then the registered owner(s) is responsible for forwarding these to the operating location(s) or person(s).

Changes in aircraft registration may take a substantial amount of time to be recorded by the Federal Aviation Administration and received by Piper to change the mailing address. Owners and operators should make arrangements to keep abreast of flight manual revisions during this interim period through their Piper Dealer/Service Center.

The Federal Aviation Administration (FAA) publishes Airworthiness Directives (AD's) that apply to specific aircraft. They are mandatory changes and are to be complied within a time limit set by the FAA. When an AD is issued, it is sent to the latest registered owner of the affected aircraft and also to subscribers of their service. The owner is solely responsible for being aware of and complying with airworthiness directives.

NOTE: A searchable database of AD's is available on the FAA website. See the "Airworthiness Directives" link at http://www.faa.gov.

Owners should periodically check with a Piper Dealer/Service Center to find out the latest information to keep his aircraft up to date.

Service Bulletins, Service Letters, and Service Spares Letters are also available by subscription. See the availability statement in Revision Status.

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TIME LIMITS

1. General

A. Refer to 4-00-00 for the FAA-approved airworthiness limitations section. It sets forth each mandatory replacement time, structural inspection interval, and related structural inspection procedure required for type certification.

B. Refer to 5-20-00 for Piper's recommended Inspection Programs. They include the frequency and extent of the inspections required for the continued airworthiness of these airplanes.

C. Inspections required by Flight Hour or Calendar Year, if due, are included as part of the Annual / 100 Hour Inspection and/or the Progressive Inspection Event cycles, and are listed individually in 5-30-00.

2. Life Limited Parts Marking and Disposition

14 CFR Part 43.10, Disposition of Life-Limited Aircraft Parts requires that proper procedures are followed when removing life limited parts with time and/or cycles remaining on them as well as the disposition of life limited parts with no time and/or cycles left. Life limited parts defined by Type Certificate (TC) are listed in 4-00-00. Other parts which are replaced or rebuilt after having accumulated cycles, hours, or other replacement limit are specified in Chapter 5.

A. Parts that are removed prior to accumulating their life limit, are to be marked with indelible ink or marker with the part number, serial number and accumulated life status as defined in 14 CFR Part 43.10 in a manner that does not affect part structural integrity, i.e. - no surface deformation such as vibration/etching allowed.

B. Parts that have accumulated the life limit shall be disposed of in accordance with the applicable FARs. Piper recommends life limited parts with no time and/or cycles remaining be completely destroyed.

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SCHEDULED MAINTENANCE

WARNING: GROUND THE MAGNETO PRIMARY CIRCUIT (P LEAD), BEFORE PERFORMING ANY MAINTENANCE OPERATION ON THE ENGINE.

1. Description

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE MAINTENANCE MANUAL - INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The recurring maintenance schedule for the PA-46-350P Malibu Mirage/M350 (S/N's 4636001 and up) and PA-46R-350T Malibu Matrix is provided herein as an Annual / 100 Hour Inspection. A Progressive Inspection Program is available in a separate manual form. See the availability statement in Revision Status.

Piper inspection programs comply with the F.A.A. Federal Aviation Regulations Parts 43, 91 and 135. The owner/operator is primarily responsible for maintaining the airplane in an airworthy condition, including compliance with all applicable Airworthiness Directives and conformity with the requirements in FAR 91.409, 91.411 and 91.413.

The first overhaul or replacement of components should be performed at the given periods. The condition of various components can then be used as criteria for determining subsequent periods applicable to the individual airplane, depending on usage, providing the owner/operator has an established Part 91 Progressive Inspection Program (see 91.409(d)) or Part 135 Approved Aircraft Inspection Program (see 135.419).

The time periods given for inspections of various components are based on average usage and environmental conditions.

NOTE: The listed inspection, overhaul and replacement schedules do not guarantee that a particular item or component will reach the listed time without malfunction. Unique operating conditions encountered by individual airplanes cannot be controlled by the manufacturer.

2. Definitions

A. Inspections - Must be performed only by persons authorized by the FAA or appropriate National Aviation Authority who are qualified on these aircraft, using acceptable methods, techniques and practices to determine physical condition and detect defects.

(1) Routine Inspection - Consists of a visual examination or check of the aircraft and its components and systems without disassembly.

NOTE: This includes examination of all fasteners for broken torque stripe. If broken torque stripe is found, loosen fastener, retorque, apply torque stripe, and safety, as required. Please report broken torque stripe via the FAA Service Difficulty Reporting (SDR) system.

(2) Detailed Inspection - Consists of a thorough examination of the appliances, the aircraft and the components and systems with such disassembly as is necessary to determine condition.

(3) Special Inspection - Involves those components, systems or structure which by their application or intended use require an inspection peculiar to, more extensive in scope or at a time period other than that which is normally accomplished during the event inspection.

B. Checks - Can be performed by pilots and/or mechanics who are qualified on this aircraft and consists of examinations in the form of comparisons with stated standards for the purpose of verifying condition, accuracy and tolerances.

C. Approved Inspection - Means a continuing airworthiness inspection of an airplane and its various component and systems at scheduled interval in accordance with procedures approved by the FAA under FAR Part 91.409(d) or Part 135.419.

D. Tests - Operation of aircraft components, appliances or systems to evaluate functional performance.

(1) Operational Test - A task to determine that an item, is fulfilling its intended purpose.The task does not require quantitative tolerances. This is a fault finding task.

(2) Functional Test - A quantitative check to determine, if one or more functions of an item performs within specified limits. This test may require the use of supplemental bench test equipment.

(3) In addition, each of the above tests must be performed by an FAA Certified Repair Station with appropriate ratings or by a Certified Mechanic who is qualified on this aircraft. The recording of the above function must be made in the permanent aircraft records by the authorized individual performing the test.

E. Bench Test - Means removal of component from the aircraft to inspect for cleanliness, impending failure, need for lubrication, repair or replacement of parts and calibration to at least the manufacturers specifications using the manufacturers recommended test equipment or standards or the equivalent.

Each bench test will be performed by a Piper Service Center, FAA Certified Repair Station with appropriate rating or by a certified mechanic. This test will be performed at the scheduled interval regardless of any bench test performed on a particular component while being repaired/overhauled before scheduled interval bench test. After the component is installed into the aircraft, an operational test of the component and its related system should be performed to ensure proper function. Serviceable parts that were issued to the component will be filed in the aircraft permanent records. The person performing the test must make appropriate entries in the aircraft's permanent maintenance record.

F. Maintenance - The word maintenance as defined by FAR Part 1, means "inspection, overhaul, repair, preservation and the replacement of parts, but excludes preventive maintenance."

G. On Condition Maintenance - A primary maintenance process having repetitive inspections or tests to determine the condition of units, systems, or portions of structures with regard to continued serviceability (corrective action is taken when required by item condition).

H. Time - as used in this manual.

(1) Time-in-service for aircraft components, unless otherwise specified, is a cumulative total of flight hours or calendar time calculated for the time a new or overhauled component was first installed in any aircraft, and including:

(a) the aircraft time that elapses from the initial installation to the first removal, if any; and,

(b) the aircraft time that elapses from each subsequent installation to each subsequent removal, if any; or,

(c) the calendar time elapsed since the installation.

NOTE: Dates stamped on individual components at the time of manufacture are typically applied to determine shelf life- i.e. the maximum time allowed from manufacture/ assembly/cure until actually installed in an aircraft and are not relevant.

Do not, however; ignore markings applied to life-limited parts when removed with time and/or cycles remaining on them.

(2) Aircraft time flight hours, or aircraft hours are the Hobbs Time shown on, or calculated from the installed Hour Meter.

3. Inspection Requirements

WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

Inspections must be accomplished by persons authorized by the FAA or appropriate National Aviation Authority. Checks may be performed by a pilot or owner who is checked out on the airplane.

A. Annual / 100 Hour Inspection

See paragraph 4.

Owners/operators may maintain the airplane solely under FAR 91.409 (a) and (b) inspection requirements. The 100 hour inspection cycle is a complete inspection of the airplane and is identical in scope to an annual inspection.

B. Progressive Inspection

The Progressive Inspection program is designed to permit the best utilization of the aircraft through the use of a planned inspection schedule. This schedule:

P/N 767-013, PA-46-350P Malibu Mirage/M350 and PA-46R-350T Malibu Matrix

is prepared in a manual form. See the availability statement in Revision Status.

Refer to Piper's Customer Service Information File P/N 1753-755 (available online, see Availability on page iii for current web address) for a checklist to ensure obtaining latest issue.

NOTE: The PA-46-350P Malibu Mirage/M350 and PA-46R-350T Malibu Matrix 50 Hour Progressive Inspection Manual (P/N 767-013) referenced above is not a stand-alone document. The Progressive Inspection Manual constitutes a snapshot of the Airworthiness Limitations and Inspection sections of the Instructions for Continued Airworthiness (ICA) and is current only at the time of printing. Use it as follows:

(a) Owners/operators desiring to establish a Part 91 Progressive Inspection Program (PIP) (see 91.409(d)) or a Part 135 Approved Aircraft Inspection Program (AAIP) (see 135.419) should use the Progressive Inspection Manual as a template for submission to the appropriate FAA office.

(b) Service centers conducting Event Cycle inspections under a FAA-approved PIP or AAIP can use the Progressive Inspection Manual as a working check-off list/form, provided they verify its currency against the FAA-approved PIP or AAIP.

C. Overlimits Inspections

If the airplane has been operated so that any of its components have exceeded their maximum operational limits, special inspections may be required by Piper and/or the component manufacturer. See 5-50-00 and applicable vendor publications.

4. Annual / 100 Hour Inspection Procedure

A. Scheduled Maintenance (i.e. - paragraph 5)

(1) The required periodic inspection procedures are listed in paragraph 5. These inspection procedures are broken down into major groups which include Propeller, Engine, Turbocharger (where applicable), Cabin and Cockpit, Fuselage and Empennage, Wing, Landing Gear, Special Inspections, Operational Inspection and General. The first column in each group lists the inspection or procedure to be performed. The second column is divided into two sub-columns indicating the required inspection interval of 50 hours or 100 hours. Each inspection or operation is required at each of the inspection intervals indicated by a circle (O). When a vendor publication specifies a time outside the 50 / 100 hour cycle, it will be listed as a special inspection in 5-30-00.

(2) Refer to the applicable chapter of this manual for instructions on how to gain access to remove any item that must be removed and is not completely accessible.

(3) Inspection Report Forms.

To help in the performance of periodic inspections, an Inspection Report form:

P/N 767-011 for the PA-46-350P Piper Malibu Mirage/M350 (S/N's 4636001 & up) and

PA-46R-350T Malibu Matrix (S/N's 4692001 & up)

is available. See the availability statement in Revision Status.

NOTE: Service centers conducting Part 91 Annual / 100 Hour Inspections can use the Inspection Report Form (P/N 767-011), as a working check-off list, provided they verify its currency against an up-to-date copy of the ICA (i.e. - this Maintenance Manual, see 4-00-00 and 5-20-00).

(4) In addition to inspection intervals required in scheduled maintenance (i.e. - paragraph 5), preflight inspection must also be performed.

(5) References to maintenance manual applicable areas are per the “chapter - system/sub-system” assignment of subject material numbering system.

B. Special Inspections (5-30-00)

(1) Per Flight Hour

(2) Per Calendar Year

(3) Per Specific Operation / Operating Environment

5. Scheduled Maintenance

Refer to Notes 1, 2, 3, and 4 before performing any of the following inspections.

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
A. PROPELLER GROUP		
WARNING: USE EXTREME CAUTION WHEN ROTATING PROPELLER BY HAND; PROPELLER MAY KICK BACK. PRIOR TO ROTATING PROPELLER ENSURE BOTH MAGNETO SWITCHES ARE OFF (GROUNDED). IF MAGNETOS ARE NOT GROUNDED, TURNING PROPELLER MAY START ENGINE.		
1. Inspect spinner and spinner bulkhead for cracks	O	O
2. Remove and inspect spinner and spinner bulkhead for cracks		O
3. Inspect blades for cracks, nicks, gouges, erosion, and evidence of strikes.	O	O
4. Inspect for grease and oil leaks.	O	O
5. Inspect propeller mounting bolts for condition and security. If safety is broken, re-torque and safety		O
6. Inspect hub parts for cracks, corrosion, and wear.	O	O
7. Inspect condition of propeller deicer system (if installed)		O
8. If installed, inspect propeller deice slip rings, brush block and brush length, replace as required		O
9. Lubricate propeller. (See appropriate Hartzell Owner's Manual and Lubrication Chart, 12-20-00.)		O
10. Install spinner		O
B. ENGINE GROUP		
WARNING: GROUND MAGNETO PRIMARY CIRCUIT BEFORE WORKING ON ENGINE.		
NOTE: Read Note 5 before completing this group.		
1. Remove engine cowling	O	O
2. Clean and check cowling for cracks, distortion, and loose or missing fasteners	O	O
3. Compression check while engine is warm		O
4. Drain oil sump while engine is warm.	O	O
5. Inspect oil temperature sender unit for leaks and security		O
6. Inspect cylinder head temperature probe and wires for security.		O
7. Inspect oil lines and fittings for leaks, security, chafing, dents, and cracks.	O	O
8. Clean and inspect oil radiator cooling fins.		O
9. Clean oil suction screen and inspect for foreign particles	O	O
10. Change full-flow (cartridge-type) oil filter element, check element for foreign particles. (See 12-10-00.)	O	O
11. Fill engine with oil. (See 12-10-00.)	O	O
CAUTION: DO NOT CONTAMINATE VACUUM PUMP WITH CLEANING FLUID.		
12. Clean engine. (See Note 6.)		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
B. ENGINE GROUP (CONT.)		
13. Inspect condition of spark plugs, clean and adjust gap as required		O
NOTE: If fouling of spark plugs is apparent, rotate bottom plugs to upper plugs.		
NOTE: Piper does not recommend that Iridium spark plugs be cleaned using abrasive or glass bead materials except as specified in the latest revision of Champion Aviation Technical Bulletin 93-2 for servicing Champion Iridium "S" spark plugs.		
14. Inspect spark plug cable, lead spring, and silicone collar for corrosion, deposits, and condition		O
15. Inspect ignition harnesses and insulators for high tension leakage and continuity		O
16. Inspect cylinders for cracked or broken fins. (See Note 7.)		O
17. Inspect rocker box covers for evidence of oil leaks, if leaks are detected replace gasket and torque cover screws to 50 in.-lbs.	O	O
18. Inspect all lines, air ducts, electrical leads (wiring) and engine attachments for general condition (chafing, cracks or cracked insulation, deterioration), security, proper routing, and correct installation	O	O
19. Inspect terminals for security and cleanliness.		O
20. Inspect magnetos for oil seal leakage		O
21. Inspect magnetos to engine timing. (See Notes 15 and 16.)		O
22. Inspect security of injector nozzles and sense line.		O
NOTE: Clean injector nozzles as required. (Clean with acetone only.)		
23. Remove air filter and clean per 12-10-00, replace as required.	O	O
24. Inspect intake ducts for leaks and all wires that form duct must be in place and secure		O
25. Inspect condition and operation of alternate air door and box		O
26. Inspect alternate air door assembly (flapper valve plate) for cracks and condition		O
27. Inspect flexible hoses for condition. (See Note 8.)	O	O
28. Inspect fuel system for leaks including flow dividers, lines, and fittings per latest revision Lycoming Service Bulletin 342, Fuel Line and Support Clamp Inspection and Installation		O
29. Inspect engine-driven and electric fuel pumps for condition and operation. Replace as required. (See 28-20-00; 73-30-00, Fuel Pump, Pressure Operational Check; and Note 10.)		O
30. Inspect condition of vacuum pumps and security of hoses. (See Notes 12 and 13.)		O
31. Inspect throttle, mixture, and propeller governor controls for travel and operating condition. (Ensure full stop to stop travel.)		O
32. Inspect mixture control cable for heat damage and routing		O
33. Inspect exhaust stacks for cracks, hot spots, and security. Inspect gaskets for leakage and condition. (Replace gaskets as required.)	O	O
34. Inspect exhaust pipe and heat exchanger. (Refer to 81-20-00, Turbocharger and Exhaust System Visual Inspection.)	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
35. Inspect Right Hand exhaust heater muff for leaks, (Refer to 81-20-00, Turbocharger and Exhaust System Visual Inspection and applicable engine manufacturer instructions.)		O
36. Inspect exhaust heat shield and cross-over tubes for cracks and condition		O
37. Inspect breather tube for obstructions, coking, and security		O
38. Inspect security and condition of oil separator		O
39. On left-hand tailpipe, disconnect breather tube hose connection. Remove lower tube assembly from tailpipe at slip joint. Remove any contaminate buildup		O
40. Inspect crankcase for cracks, leaks, and security of seam bolts		O
41. Inspect engine mount for cracks, corrosion, and loose mounting bolts. (See Note 14.)		O
42. Inspect engine mount heat shield for cracks, corrosion and condition.		O
43. Inspect all engine baffles for cracks and security		O
44. Inspect rubber engine shock (isolator) mounts for deterioration. Replace as required		O
45. Inspect firewall for cracks, condition, and security		O
46. Inspect condition of firewall sealing		O
47. Inspect alternator and mounts for cracks, condition, and security		O
48. Inspect condition and tension of alternator drive belts (Refer to 24-30-00.)		O
49. Inspect starter and mounts for cracks, condition, and security		O
50. Inspect air conditioning system for evidence of refrigerant leakage. (Refer to 21-50-00.)		O

NOTE: If cooling system has leaked refrigerant or is discharged, the compressor oil level must be Inspected. Environmental regulations may prohibit adding refrigerant until the cause of the loss has been determined and repaired.

51. Inspect condition and tension of compressor drive belt. (Refer to 21-50-00.)		O
52. Inspect security of compressor mounting		O
53. Inspect compressor clutch security and condition of wiring		O
54. Inspect engine accessory case and components for leakage, condition, and security		O
55. Inspect bleed air ducts for condition and security		O
56. Inspect sonic nozzles for condition and security		O

CAUTION: DO NOT LUBRICATE TEFLON LINERS IN CONTROL CABLES.

57. Lubricate all controls per Lubrication Chart, 12-20-00		O

C. TURBOCHARGER GROUP

1. Visually inspect system for oil leaks, exhaust system leaks and general condition. (See 81-20-00, Turbocharger and Exhaust System Visual Inspection.)	O	O
2. Inspect the compressor wheel for nicks, cracks, or broken blades		O
3. Inspect for excess bearing drag or wheel rubbing against housing		O
4. Inspect turbine wheel for broken blades or signs of rubbing		O
5. Inspect oil inlet and outlet ports in center housing for leaks		O
6. Inspect turbo gaskets for leaks	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
C. TURBOCHARGER GROUP (CONT.)		
7. Inspect turbo clamp for cracks and torque. (If the clamp nut is removed, replace with a new nut (P/N 755-657). Torque the nut on installation. Apply torque stripe and inspect after first engine run.)	O	O
8. Inspect mounts for cracks, corrosion, clearance, and security		O
9. Inspect wastegate actuator linkage, rod ends, springs, butterfly, and bushings for condition. Maintain wastegate per the component manufacturer's instructions. Replace as required		O
10. Inspect drain line from actuator for presence of oil	O	O
11. Inspect induction and exhaust components for worn or damaged areas, loose clamps, cracks, and leaks.	O	O
12. Perform V-Band Coupling 100 Hour Inspection, 81-20-00		O
13. Inspect intercooler clips for condition and security.	O	O
14. Inspect turbo and controller system oil lines for leaks and security	O	O
15. Inspect for oil leakage from controller and sense lines	O	O
16. PA-46-350P only, inspect flexible bleed air lines (high temp) for condition and security. Replace on condition.		O
17. Install engine cowling	O	O
D. CABIN AND COCKPIT GROUP		
1. Remove inspection plates and panels (seats and carpet must be removed for access to inspection plates on cabin floor). (See Figure 3, 6-00-00.)	O	O
2. Inspect cabin entrance door and emergency exit door seals		O
3. Inspect cabin door and emergency exit for proper rigging, and Inspect retainer pins and striker plates for bending, cracks, proper engagement, pulled or sheared fasteners and bending of the frame web		O
4. Perform functional test of emergency exit. Verify that no exceptional effort is required to remove Emergency Exit Door. Remove and reinstall Emergency Exit Door.		O
CAUTION: WITH CREW SEATS OCCUPIED AND AT THE LOWEST VERTICAL POSITION, INSPECT SEATS FOR PROPER ENGAGEMENT OF PINS TO SEAT TRACK.		
5. Inspect crew seats for proper vertical and horizontal operation	O	O
6. Inspect seats and attaching brackets and hardware for condition, security, and operation		O
7. Inspect seat belts and shoulder harnesses per 25-10-00, Restraint System		O
8. Inspect electric trim operation and indicators for full travel, binding, damage, and correct control deflection		O
9. Inspect rudder pedals and toe brakes for travel, binding, and security		O
10. Ensure no hose clamps or cable ties are attached to the torque tubes of the rudder pedal and bar assembly. Verify free and correct movement of the rudder pedal and bar assembly		O
11. Inspect parking brake for condition, security and operation		O
12. Inspect control wheels, column, pulleys, cables, turnbuckles and fittings for condition, security, and full travel. Inspect cables per Control Cable Inspection, 27-00-00		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
13. Inspect push pull tubes, torque tubes, levers, pillow blocks, bellcranks, and connections for condition and security		O
14. Check operation of strobe, landing, navigation, cabin, and instrument lights	O	O
15. Inspect condition of instruments, lines, hoses, and attachments		O
16. In airplanes equipped with Garmin G1000/G1000 NXi, perform 100 hour inspections listed under Inspections in 34-25-01 or 34-25-02, as appropriate		O
17. Inspect pneumatic and electric gyro instruments (overhaul or replace as required)		O
18. If installed, replace vacuum regulator filter element		O
19. Inspect pitot/static and alternate static lines, hoses for condition and security		O
20. Inspect static system, altimeter and transponder for installation/certification per latest revision of AC43.13-1 and current test/inspection per FAR's 91.411 and 91.413, respectively		O
21. Inspect operation and security of fuel selector valve linkage at fuel selector	O	O
22. Inspect condition of safety wire on screw attaching valve cam		O
23. Check operation of cabin heat and defrost controls		O
24. Inspect all knobs, switches, and levers for security of attachment and condition		O
25. Inspect condition of environmental system ducts		O

WARNING: BE SURE SKIN AND CLOTHING ARE FREE OF GREASE, OIL, OR OTHER PETROLEUM PRODUCTS BEFORE INSPECTING OR PERFORMING MAINTENANCE ON ANY COMPONENT OF THE OXYGEN SYSTEM.

NATURE OF INSPECTION	50	100
26. Inspect crew and passenger oxygen systems. PA-46-350P only, see 100 Hour Inspection, Fixed Oxygen Generator System, 35-20-00		O
27. Inspect flap and main landing gear switches for operation, security, and condition.		O
28. Inspect all fluid lines for leakage, condition, and security. (See Note 8.)	O	O
29. Inspect electrical panel components and circuit breakers for condition and security of installation	O	O
30. Cycle each circuit breaker with airplane power off	O	O
31. In PA-46-350P S/N's 4636001–4636651 less 4636633: Inspect forward side of outflow and safety valves for condition and security. Remove and clean per 21-30-00.		O

NOTE: In PA-46-350P S/N's 4636001–4636651 less 4636633: Outflow and safety valves may require cleaning prior to 100 hour intervals, if heavy smoke or dusty conditions exist.

NATURE OF INSPECTION	50	100
32. In PA-46-350P S/N's 4636001–4636651 less 4636633: Inspect filters on controller and safety valve for contamination. Clean or replace as required. (See 21-30-00.)		O
33. In PA-46-350P S/N's 4636633, 4636652 and up: Inspect forward side of outflow valves for condition and security.		O
34. In PA-46-350P only, inspect aft face of F.S. 100.00 bulkhead and forward face of F.S. 273.746 bulkhead for bulging, cracks, dents, loose, or missing fasteners, condition and security of components.		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
D. CABIN AND COCKPIT GROUP (CONT.)		
35. If installed, inspect disposable-type (non-gauged) fire extinguisher minimum weight as specified on nameplate		O
36. Inspect avionics compartment, components, and wiring for condition, security, and operation.		O
37. Inspect control cable boots for condition and security.		O
38. Inspect air conditioning evaporators (if installed)		O
39. Lubricate per Lubrication Chart, 12-20-00		O
40. Install inspection plates and panels	O	O
E. FUSELAGE AND EMPENNAGE GROUP		
1. Remove inspection plates and panels. (See Figure 1, 6-40-00.)	O	O
2. Inspect skins, bulkheads, frames, and stringers for damage, irregularities, or structural defects (i.e., skin cracks, distortion, dents, corrosion, and loose or missing rivets)		O
3. Inspect all fuselage and empennage drain holes. Ensure they are clear and open		O
CAUTION: SEE LIMITATIONS, 56-00-00.		
4. Inspect windshield and windows for nicks, scratches, cracks, crazing, and discoloration. (See 56-00-00.)		O
5. Inspect avionics compartment, components, and wiring for condition, security, and operation.		O
6. Inspect control cable boots for condition and security.		O
7. Check fuel sump drains for water and proper operation	O	O
8. Drain static lines. (Refer to 34-10-00.).		O
9. Inspect antennas/coaxial cables for condition and security		O
10. Inspect dorsal fin for condition and security		O
11. Inspect rudder for surface damage or irregularities (i.e., skin cracks, distortion, dents, and corrosion); structural defects (i.e., loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
12. Inspect rudder hinges and attachments for damage and operation		O
13. Inspect rudder hinge bolts for excessive wear		O
14. Inspect rudder trim mechanism installation.		O
15. Inspect vertical fin for surface damage or irregularities (i.e., skin cracks, distortion, dents, and corrosion); structural defects (i.e., loose or missing rivets); and attachment points for missing or worn hardware		O
16. Inspect vertical fin attach points and wiring.		O
17. Inspect forward vertical fin attach fittings for corrosion and damage. (See 55-30-00, Attach Fittings Corrosion Control.)		O
18. Inspect condition of deice system (if installed)		O
19. Inspect elevator for surface damage or irregularities (i.e., skin cracks, distortion, dents, and corrosion); structural defects (i.e., loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
20. Inspect elevator hinge bolts, trim tab hinges, and attachments for damage and operation. (See 55-20-00, Elevator Trim Tab Rods Corrosion Control.)		O
21. Inspect elevator and trim tab hinge bolts and bearings for excessive wear		O
22. Inspect elevator plate assembly rivets per 55-20-00.		O
23. Inspect horizontal stabilizer for surface damage or irregularities (i.e., skin cracks, distortion, dents, and corrosion); structural defects (i.e., loose or missing rivets); misrigging; and attachment points for missing or worn hardware		O
24. Inspect horizontal stabilizer attachments		O
25. Inspect forward and aft horizontal stabilizer attach fittings for corrosion and damage. (See 55-10-00, Attach Fittings Corrosion Control)		O
26. Inspect elevator trim mechanism installation.		O
27. Inspect battery, battery compartment, and vent system for water, corrosion, etching, condition, and security. Flush compartment as required		O
28. Inspect external power supply receptacle, battery relay, fuses, and vent blower assembly for cleanliness, corrosion, condition, and security		O
29. Inspect fuel filter. Drain and flush completely. Clean or replace as required	O	O
30. Inspect deice system pneumatic valves and lines for condition and security (if installed)		O
31. Inspect baggage compartment door, latches, and hinge for operation, condition, and security		O
32. Inspect baggage compartment upholstery for condition		O
33. Inspect baggage compartment light and switch for operation, condition, and security.		O
34. Inspect ELT battery mount for condition and security		O
35. Inspect ELT antenna for condition, security, and operation (Inspect operation per Antenna Test in 25-60-00). Replace antenna if bent or damaged		O
36. Inspect fluid level in hydraulic reservoir. Fill as required	O	O
37. Inspect hydraulic power pack for condition, leaks, and security		O
38. Inspect all fluid lines for leakage, condition, and security. (See Note 8.)		O
39. Inspect autopilot servos and controls, bridle cables, terminals, turnbuckles, fittings, guides, and pulleys for safeties, condition and wear, proper installation and function. Inspect bridle cable tension. (See 22-10-00.) Inspect condition of cables per Control Cable Inspection, 27-00-00		O
40. In airplanes equipped with Garmin GFC 700 and GSM-85A servo gearboxes, remove Pitch, Pitch Trim, Roll, and Yaw Servos and inspect slip clutch torque per Slip Clutch Torque Adjustment under Garmin GFC 700 in 22-10-00.		O
41. Inspect aileron, rudder, elevator, and elevator trim cables and pulleys for safety, condition, and operation. Inspect cable terminals, turnbuckles, guides, and fittings for safety and condition. Inspect per Control Cable Inspection, 27-00-00		O
42. Inspect rudder, elevator, and elevator trim cable tension per 27-00-00, Chart 2. Use a tensiometer.		O
43. Conduct a general visual inspection of electrical and electronic installations (mounting, wiring, harnesses, shields, connectors, etc.) for condition and security		O
44. In PA-46R-350Ts and PA-46-350Ps (S/N's 4636375 and up), visually inspect the lightning diverter per 51-80-00, Lightning Diverter, Inspection		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
E. FUSELAGE AND EMPENNAGE GROUP (CONT.)		
45. Inspect all non-flight control cables, air ducts, electrical leads, harnesses, and attaching parts for security, routing, chafing, deterioration, wear and correct installation. (Complete electrical bonding inspections per 51-80-00, Inspections, 100 Hour.)		O
46. Inspect heater ducting for obstruction, condition, and security		O
47. Inspect air conditioning evaporators and clean fins and filters		O
48. If installed, inspect electric windshield for general condition, correct fit, distortion, cracking, delamination, security, and proper operation		O
CAUTION: THE FLUX DETECTOR IS SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.		
49. Inspect flux detector		O
50. In PA-46-350P S/N's 4636001–4636651 less 4636633: Inspect aft side of outflow and safety valves for condition and security		O
51. In PA-46-350P S/N's 4636633, 4636652 and up: Inspect aft side of outflow valves for condition and security		O
52. Inspect all hoses and lines for leakage, condition, and security. (See Note 8.)		O
53. Lubricate per Lubrication Charts, 12-20-00		O
54. Install inspection plates and panels	O	O
NOTE: Ensure correct hardware is used when reinstalling vertical fin dorsal fairing over flux detector.		
55. Inspect static wicks per Static Wicks, Inspection, 23-60-00. (Replace as required.)		O
F. WING GROUP		
1. Remove inspection plates and panels. (See Figure 3, 6-00-00.)	O	O
2. Inspect exterior surfaces, skins, and tips for damage and loose or missing fasteners	O	O
3. Inspect visible interior structural components (skins, spars, ribs, stringers, etc.) for condition, security, distortion, or failure		O
4. Inspect wing carry-through structure for condition, security, distortion or failure		O
5. Inspect forward and aft wing spar to fuselage attach fittings for corrosion and condition. (See 57-40-00, Attach Fittings Corrosion Control.)		O
6. Inspect ailerons for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excess wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
7. Inspect aileron bellcranks for damage and operation		O
8. Inspect aileron cables, pulleys, and bellcranks for safety, condition, and operation. Inspect aileron cable terminals, turnbuckles, fittings, and guides for safety and condition. Inspect per Control Cable Inspection, 27-00-00		O
9. Inspect aileron cable tension per 27-00-00, Chart 2. Use a tensiometer		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
10. Inspect flaps for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge or track damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
11. Inspect the flap actuation system rod end bearings for binding or seizing of the ball bearings. Pay particular attention to the rod end bearings attached to the inboard bellcrank assembly located in each main gear well		O
12. Inspect static wicks per Static Wicks, Inspection, 23-60-00. (Replace as required.)		O
WARNING: PHYSICAL MANIPULATION OF LIFT TRANSDUCER ON THESE AIRPLANES MAY RESULT IN ERRONEOUS STALL WARNINGS NECESSITATING CALIBRATION OF SYSTEM. SEE 27-30-00.		
13. Inspect lift transducer for security of attachment, spring centering of switch blade, and corrosion. Inspect heat element for operation.		O
14. Inspect wing inspection light for broken lens, security, wiring, and attachment (if installed)		O
15. Lubricate per Lubrication Chart, 12-20-00		O
16. Inspect condition of hydraulic, fuel, and pneumatic hoses and lines. (Refer to Note 8.)		O
17. Inspect condition and security of wiring and connectors		O
18. Inspect fuel tanks and lines for leaks		O
NOTE: Any fuel leaks within the wing will show at the lower wing root fairing, between wing and fuselage.		
19. Inspect fuel filler cap, O-ring, and receptacle for condition	O	O
20. Inspect fuel tanks marked for proper octane rating	O	O
21. Inspect fuel caps, cap gaskets, fuel filler neck, and gage transmitter access covers, for condition, proper sealing, security, alignment, etc. Ensure to service and clean these areas, replacing parts as necessary		O
22. Inspect the interior of metal fuel tanks for signs of corrosion, which may indicate water contamination. If signs of contamination are found, alert the owner and fuel supplier of your findings for corrective action		O
23. Inspect radome for erosion, cracks, or delamination (if installed)		O
24. PA-46-350P only (S/N's 4636375–4636651, less 4636633): inspect standby attitude indicator emergency power supply (in radar pod aft compartment) for condition and security.	O	O
25. Inspect condition of pneumatic deicer (if installed)		O
26. Inspect wing tip navigation lights for broken lenses, security, wiring, and attachment	O	O
27. If installed, inspect SpeedBrakes per 50 Hour Inspection, 27-60-00	O	O
28. Install inspection plates and panels	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
G. LANDING GEAR GROUP		
1. Inspect oleo struts for proper extension and evidence of fluid leakage. (See Landing Gear, 12-10-00.)	O	O
CAUTION: WHEN AIRCRAFT HAS OPTIONAL RADAR POD INSTALLED ON RIGHT WING, ENSURE THAT JACK DOES NOT COME INTO CONTACT WITH RADAR POD.		
2. Place airplane on jacks. (Refer to 7-10-00.)		O
3. Remove inspection covers and panels		O
4. Inspect nose gear steering control and travel		O
5. Inspect nose gear installation for condition and security		O
6. Inspect nose gear trunnion for cracks and condition	O	O
7. Inspect nose wheel steering cam and rotator for cracks		O
8. Inspect squat switch for security and adjustment		O
9. Inspect tires for cuts, uneven or excessive wear, and slippage	O	O
10. Remove wheels and clean, inspect and repack bearings		O
11. Inspect wheels for cracks, corrosion, and broken bolts		O
12. Inspect brake linings and discs for condition and wear		O
13. Check tires for proper pressure. (Refer to 12-10-00, Tires and 6-00-00, Chart 1.)	O	O
14. Inspect brake hydraulic lines for leakage, condition, and security		O
15. Inspect hydraulic lines, electrical leads, and attaching parts for condition and security (i.e. - routing, chafing, damage, wear, etc.)		O
16. Inspect brake reservoir for proper fluid level, leaks, condition, and security	O	O
17. Inspect gear fork for damage		O
18. Inspect struts for fluid/pressure leaks and scoring		O
19. Inspect torque links, bolts, and bushings. Rebush as required		O
20. Inspect gear struts, trunnion pins and attachments for condition and security		O
21. Inspect bolt, bushings, trunnion pins, and attachments for condition and security. Rebush as required		O
22. Inspect retraction actuators and attachments for condition and security		O
23. Inspect condition and security of flexible hydraulic lines to actuator. (Replace flexible hoses as required, see Note 8.)		O
24. Inspect main and nose gear doors, and rod assemblies for corrosion, security, and freedom of movement		O
25. Inspect locking actuator for operation and adjustment		O
26. Inspect nose gear door actuator for operation and adjustment		O
27. Lubricate per Lubrication Chart, 12-20-00		O
28. Inspect wiring and wiring harness for condition, security, damage, chafing, and corrosion. Inspect switches for adjustment, security and operation		O
29. Inspect actuating cylinders for leakage and security		O
30. Perform Landing Gear Retraction System Functional Test, 32-30-00		O
31. Inspect the free fall valve and associated lines and fittings for leakage. Replace on condition.		O
32. Place weight of aircraft on landing gear. (DO NOT REMOVE JACKS.)		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
33. Inspect anti-retraction system on gear lever for proper operation		O
34. Install inspection covers and panels		O
WARNING: DO NOT REMOVE JACKS UNTIL IT HAS BEEN DETERMINED THAT THE LANDING GEAR IS DOWN AND LOCKED AND ANTI-RETRACTION SYSTEM HAS BEEN INSPECTED.		
35. Ensure all gears are down and locked, then remove jacks		O
36. Inspect wheel alignment		O
H. SPECIAL INSPECTIONS		
Review inspections in 5-30-00. Perform all special inspections applicable to your aircraft and currently due per the given inspection interval	O	O
I. OPERATIONAL INSPECTION		
- ENGINE OFF		
1. Inspect electronic equipment operation	O	O
2. Inspect free and correct operation of flight controls	O	O
3. Inspect operation of flaps	O	O
4. If installed, inspect and self-test Transicoil EMIS Instrument Stack. (See 77-40-00, Dual Analog Module Indicator, Performance.)	O	O
5. In PA-46-350P S/N's 4636021 and up and PA-46R-350Ts, inspect manifold pressure transducer per the Basic Test under Manifold Pressure Transducer Installation, 77-10-00	O	O
6. Inspect communication receive capability using Failsafe Path and COM1:		
(a) If equipped with GMA-340, see 23-50-00, GMA-340, System Functional Test, Failsafe Operation Check.		O
(b) If equipped with GMA-347 or GMA-350(c), see GMA-347 Test or GMA 350(c) Test in 34-25-01 or 34-25-02, as appropriate		O
- ENGINE RUNNING		
7. Inspect fuel pump and fuel tank selector	O	O
8. Inspect fuel quantity and flow gauges	O	O
9. Inspect oil pressure and temperature	O	O
10. Inspect manifold pressure	O	O
11. Inspect tachometer	O	O
12. Inspect alternator output on #1 alternator and #2 alternator	O	O
13. Inspect parking brake and toe brakes	O	O
14. Inspect vacuum gauge display and vacuum pump operation	O	O
15. Inspect gyros for noise and roughness	O	O
16. Inspect cabin heat and defroster operation	O	O
17. Inspect magneto switch operation	O	O
18. Inspect magneto RPM variation	O	O
19. Inspect throttle and mixture operation	O	O
20. Inspect propeller smoothness	O	O
21. Inspect constant speed propeller action	O	O
22. Inspect engine idle	O	O
23. Inspect alternate air	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
I. OPERATIONAL INSPECTION (CONT.)		
24. Inspect air conditioning compressor clutch operation	O	O
25. PA-46-350P only, inspect cabin pressure controller. (See Operational Inspection, Cabin Pressure Controller, 21-30-00.)		O
J. GENERAL		
1. Aircraft conforms to FAA Specifications	O	O
2. Latest revision of applicable FAA Airworthiness Directives complied with	O	O
3. Current and correct Pilot's Operating Handbook (POH) is in the airplane	O	O
4. Inspect airplane for required placards as specified in Section 2 of the POH		O
5. Appropriate entries made in the Aircraft and Engine Log books	O	O
6. Airworthiness and Registration Certificates in the aircraft and properly displayed	O	O
7. Aircraft Equipment List, Weight and Balance and FAA Form(s) 337 (if applicable) are in the aircraft and in proper order	O	O
8. Operational inspection and run-up completed	O	O
9. Aircraft cleaned and lubricated after wash (as required)	O	O

K. NOTES

1. Refer to Piper's Customer Service Information File P/N 1753-755 (available online at http://pubs.piper.com). for latest revision dates to Piper Inspection Manuals and this maintenance manual. References to Chapter or Section are to the appropriate Chapter or Section in this manual.

   WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

2. Inspections or operations are to be performed as indicated by a "O" at the 50 to 100 hour inspection interval. Inspections or operations (i.e., component overhauls/replacements, etc.) required outside the 50/100 hour cycle are listed as special inspections in 5-30-00. Inspections must be accomplished by persons authorized by the FAA or appropriate National Aviation Authority. Checks may be performed by a pilot or owner who is checked out on the airplane.
   (a) The 50 hour inspection accomplishes preventative maintenance, lubrication and servicing as well as inspecting critical components.
   (b) The 100 hour inspection is a complete inspection of the airplane, identical to an annual inspection.

   NOTE: A log book entry should be made upon completion of any inspections.

3. Piper Service Bulletins are of special importance and Piper considers compliance mandatory. In all cases, see Service Bulletin/Service Letter Index P/N 762-332 to verify latest revision, available online, see Revision Status for current web address. See also "Chart 1" on page 052019.
4. Piper Service Letters are product improvements and service hints pertaining to servicing the airplane and should be given careful attention.
5. Inspections given for the power plant are based on the engine manufacturer's operator's manual (Lycoming P/N 60297-27) for this airplane. Any changes issued to the engine manufacturer's operator's manual shall supersede or supplement the inspections outlined in this report. Should fuel other than the specified octane rating for the power plant be used, refer to the latest revision of Lycoming Service Letter No. L185 for additional information and recommended service procedures.
6. Cover or remove all filters, plug all openings, cover the ignition lead at the spark plug and magneto. Solvent will contaminate the air and fuel filters and will corrode the springs in the ignition leads.
7. Check cylinders for evidence of excessive heat which is indicated by burned paint on the cylinders. This condition is indicative of internal damage to the cylinder, and if found, its cause must be determined and corrected before the airplane is returned to service.

   Heavy discoloration and the appearance of seepage at the cylinder head and barrel attachment area are usually due to emission of thread lubricant used during assembly of the barrel at the factory, or by slight gas leakage which stops after the cylinder has been in service for a while. This condition is neither harmful nor detrimental to engine performance and operation. If it can be proven that leakage exceeds these conditions, the cylinder should be replaced.

K. NOTES (CONT.)

8. Flexible hose replacement times are in-service times. In-service dates must be determined by (1) the date the aircraft was licensed, if new or (2) the date entered in the logbook for the replacement hose placed in service. Do not use the date stamped on the hose, as time may be included for shelf life, and not in-service use.
9. Not used.
10. In S/N's 4636001 thru 4636221 only, in airplanes which have not installed a Lear Romec P/N RG9080-J4A/M engine-driven fuel pump, verify compliance with latest revisions of Piper Service Bulletin No. 1035A, Lycoming Service Bulletin No. 529B, and Crane/Lear Romec Service Bulletin No. 101SB020. See also FAA AD 2003-18-12.
11. Not used.
12. For airplanes equipped with Parker Hannifin / Airborne vacuum pump(s), verify compliance with Parker Hannifin / Airborne Service Letter No. 72.
13. For airplanes equipped with Aero Accessories, Inc. Tempest Dry Air Pumps with Wear Indicator Ports, beginning at 300 hours time-in-service, and each 100 hours thereafter, inspect per Vacuum Pump(s), Inspection, 37-10-00.

    NOTE: For airplanes with frequent deice cycles, begin this inspection at 200 hours time-in-service.

14. For any airplane with engine mount P/N 89137-041 or P/N 89137-042 installed: for engine mounts which have accumulated 200 hours time-in-service, each one hundred (100) hours time-in-service perform the 100 Hour Engine Mount Inspection in 71-20-00.
15. For high altitude operations (12,000 feet and up), more frequent ignition system maintenance is required. See Champion Aerospace / Slick SB1-88B.
16. Use of magneto manufacturer's service publication is recommended. For airplanes equipped with Slick Magnetos: inspect magneto(s) per the appropriate 100 Hour Inspection in the Slick F1100 Master Service Manual.

6. Service Publications List

"Chart 1" is a cumulative list of Piper service publications (i.e., Service Bulletins, Service Letters, etc.) applicable to the airplane models covered by this manual, with the following exceptions:

A. The following service publications have been incorporated into this manual and are not listed below: Service Bulletins 995C (Part III), 1012, 1035A, 1048, 1103F, 1175B, 1204, 1237, and 1280A; Service Letters 1041, 1052, 1053, 1069, 1074, 1085, 1086, 1091, 1097, 1100, 1106A, 1108, 1131A, 1159B, 1116, 1183, 1184, 1204, 1222B, 1261, 1265 1274, 1280, 1285A, and 1286A; Vendor Service Publications 171, 204, and 244A.

B. Nor are service publications which have become obsolete.

Kits are listed when installation of that single kit indicates compliance with the associated service publication. Kits listed may be no longer available or may have been replaced.

Effectivity is listed by airplane model and year. See the individual service publication for specific serial number applicability.

CHART 1 (Sheet 1 of 7)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-46-350P	1995	SB 990		Pacific Scientific Seat Belt Replacement
		SB 993	766-613	KLN-90B GSP Wiring Modification
	1995-1996	SB 964B		Hydraulic Power Pak Pressure Switch Replacement
		SB 994		Intermittent Main Landing Gear Indication
		SB 999		Inertia Reel Shoulder Harness Mounting Brackets
		SB 1000		Engine Electrical Harness - Chafing Protection
	1995-1997	SB 1001		Nose Gear Steering Arm Rod End - Washers
		SB 1014		Replacement of Battery Power Fuse
		SB 1017		Alternator Regulator Wiring Modification
		SB 1021	766-654	Control Wheel Quadrant Stop - Fwd Pressure Bulkhead
	1995-1998	SB 1020		ARTEX 110-4 ELT and Attachment Bracket
		SB 1027	766-656	Forward and Aft Wing Attached Fittings
	1995-1999	SB 1029		Turbocharger Oil Drain Tank Inspection/Modification
		SB 1034		Battery Master contactor Wiring Modification
	1995-2000	SL 1018A		Improved Intercooler Attachment Clips
		SB 1041		Airborne Air Filter Elements Inspection/Cleaning
		SB 1054		Pilot and Co-Pilot Seat Inspection/Modification
		SB 1060		Brake Pedal Binding
		SB 1062		Inboard Flap Drive Bellcrank Inspection/Replacement
		SB 1075A		Steering Horn Modification
		SB 1087B		Nose Gear Emergency Downspring Mechanism
		VSP 127		Lycoming SA-59-800 Connecting Rod Bearing
	1995-2002	SL 1057	767-352 & 767-353	Upper Wing Skin Inspection/Reinforcement
		VSP 142		Garmin SB-0204 GPS Software
	1995-2003	SB 1134		Servo Metered Fuel Pressure Port Leakage
		SB 1140		Oxygen Generator Protective Cap Removal

CHART 1 (Sheet 2 of 7)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-46-350P (Cont.)	1995-2005	SB 1180A		V-Band Exhaust Coupling - Turbocharged Engines
		SB 995C (Parts I & II)		Turbine Inlet Temperature (T.I.T.) System Calibration and Probe Replacement
	1995-2007	VSP 184		Sky-Tec SB 07-01 PM Model Starter Replacement
	1995-2008	SB 1190B		Aileron Quadrant Inpsection
		SB 1192	88452-002	Stall Warning Heat Cotnrol Modification
		SB 1193		Nose Gear Down Spring Replacement
		SB 1196		Collector Tank Fuel Drain Inspection
		SB 2000		Current Limiter Inspection
		SL 1130	88423-002	Lower Entry Door Upgrade
	1995-2009	SB 1271A		Wing Spar Inspection
	1995-2010	SB 1217A	88465-002	Starter Contactor Replacement
		SL 1136A		Fuel Pressure Switch Adjustment
	1995-2011	SL 1138		Safe Flight Service Letter No. 65, Lift Transducer Modification
		SL 1157		Exterior Lighting - LED Upgrade
		VSP 215		Fuel Drain Valve Adapter Assembly
	1995-2012	SL 1128		Improved Capacity Battery Replacement
		SL 1160		Interior Lighting - LED Upgrade
	1995-2013	SB 1258		Fuel Vent Valve Inspection
	1995-2016	SB 1293C	88584-701 88582-701 88582-702	Engine Mount Doubler Installation and Towing Placards Replacement
		SL 1202		Main Landing Gear Actuator Hydraulic Hose Replacement
		VSP 249		Hartzell Engine Alternator Through Bolt Replacement
	1995-2017	SB 1306	88609-001 88609-002	Wing Rib Modification
	1995-2018	SB 1327	88630-701	Aileron Control Cable Protection Tube Inspection
	1995-2019	SB 1338		Main Gear Actuator Attach Fitting Inspection
	1995-2020	SL 1233		Wing and Flap Protective Tape Improvement
	1995-up	SB 1256		Nose Landing Gear Torque Link Hardware Replacement
		SL 578B		Fuel Line and Support Clamp Inspection and Installation
		SL 1039A		Alternate T.I.T. Probe Availability (AD2011-06-10)
		SL 1093		Electrical Contactor Replacement Kit
		SL 1132		Fuel Tank Leaks
		SL 1135		Control Cable / Pulley Inspections
		SL 1159		Fuel Sender Upgrade
		SL 1224		Decorative Window Sealant Improvement
		SL 1249		Control Column Plug Modification
		SSL 437D		Aircraft Tire Options
		VSP 116B		Lycoming SI-1492D Piston Plug Wear Inspection
		VSP 148		Lycoming SB-554 & Supp 1 Crankshaft Gear
		VSP 173		Hartzell HC-SB-61-284 Compression Spring
		VSP 174		Lycoming SB-569 Crankshaft Retirement
		VSP 196		Slick SB2-08A Magneto Inspections Models 4300/6300

CHART 1 (Sheet 3 of 7)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-46-350P	1995-up	VSP 197A		Slick SB3-08A Magneto Insp.4200/6200 & 4300/6300
(Cont.)	(Cont.)	VSP 207		HET Turbocharger CHRA
		VSP 255A		AVStar Flow Divider (Lycoming Fuel Manifold) Inspection
		VSP 260		Lycoming Identification of Connecting Rods with Non-Conforming Small End Bushings
		VSP 266		Lycoming Recommended Engine Procedures for Purge of Vapor During Ground Operations
	1996-1999	SL 1023		Improved Security of the Pressure Control Cable
	1996-2000	SB 1066	767-305	Cabin Door Latch Mechanism
	1997-up	SL 1026	766-659	Storm Window Seal Improvement
	1998	VSP 175		S-Tec SB-06-001 Pitch Servo Inspection
	1998-1999	SL 1022	766-658	Lower Cowling Assembly Modification
	1998-2002	SB 1135		GMA-340 Audio Panel Wiring
	1998-2003	SB 1139A		Control Wheel Attachment Inspection
	1998-up	VSP 259		Hartzell Blade - Secondary Windings
	1999-2008	SL 1170	88542-002 88542-003	Evaporator Module Assembly Replacement Requirements
	1999-2009	SB 1181B		Flap Track/Splice Strap Inspection/Modification
	1999-2010	SL 1120		Air Conditioning Servicing
		SL 1129		Improved Landing Gear Service Replacement Actuators Available
	2000	SB 1046		POH Supplement Installation
		SB 1067A	767-301	Entertainment Console Voltage Dropping
	2000-2014	SL 1196A		Availability of GTX 33ES Transponder
	2000-2016	SB 1288A		Aft Vertical Fin Spar Inspection
	2002	SB 1108	767-335	Flux Detector Relocation
	2002-2003	SB 1126A	767-364	Emergency Gear Extension Cable Bracket Modification
	2002-2005	SB 1168	767-544	Elevator Trim Tab Hinge Rivets
	2002-2012	SB 1243C		Potential Landing Gear Malfunction
		SL 1148		G1000 Software Version 720.14
	2002-up	SL 1222		Hyddraulic Pump Pressure Switch Replacement
	2003-2006	SB 1176		Rudder Quadrant / Torque Tube Inspection - Tailcone
	2003-2007	SB 1182		Wing Inspection Light
	2004	SB 1151E		Steel Hardness Discrepancy
	2004-2008	SL 1113	88444-002	Crew Seat Back Modification
	2004-2016	SB 1290B		Flap Bracket Inspection, Inboard Bellcrank
	2004-up	SL 1089		Aircraft Painting Requirement
	2006	SB 1178		Nose Gear Forging Inspection / Replacement
		SA 46-16		CHT and TIT Probe Shield Grounds
	2006-2007	SB 1185		Placard Replacement
		SB 1187		Voltage Suppressor Replacement
	2006-2008	SL 1111	88438-002	Autopilot Pitch Servo and Programmer
		SL 1126A		Voltage Suppressor Advisory

CHART 1 (Sheet 4 of 7)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-46-350P (Cont.)	2006-2009	SB 1195B		Comm1 and Comm2 Antenna and Skin Inspection (Initial compliance only)
		VSP 202		Lycoming SB 590A Use of Aircraft Boost Pump with Kelly Aerospace 200F Series Fuel Pumps
	2006-2010	SB 1208		Lightning Diverter Inspection
		SB 1212		Inspection of Flap Position Indicator Placard
	2006-up	SL 1123		Avidyne EXP5000 Software Upgrade (Mod 57)
		VSP 182		Avidyne SB-601-00006-067 Avidyne PFD
		VSP 183		Avidyne SB-601-00006-075 PFD & Garmin w/WAAS
		VSP 191		Avidyne SA-08-001 Avidyne PFD
	2007-2009	SL 1118		Sky-Tec Starter Replacement
	2008-2010	SL 1127		Emergency Exit Interior Panel Hook and Loop Fastener Replacement
	2008-2011	VSP 217		Non-Conforming V-Band Clamps
	2008-2016	SB 1291	88581-701 88581-702	Executive Table Placards Replacement
	2008-up	VSP 219		G1000 Equipped - LNAV/VNAV Full Scale Deflection
	2009-2010	SL 1122		G1000 Split Communciations Enable
	2010	SB 1223		Wire Screen Replacement
	2010-2011	SB 1232	100700-014	Cabin Door Handle Placard Inspection
		SL 1140		G1000 Software Version 720.12
	2010-2013	SL 1168		False Alternator Fail Annunciations
	2010-2014	SB 1269A		GWX 68 Ground Rewire
		SL 1179		Garmin G1000 Software Ver. 720.15
		SL 1234		Garmin G1000 GIA Software Update for Transponder Compatibility with ADS-B Out
	2010-2015	SL 1203A		Garmin G1000 Software 720.17 with Stall Heat Control
		SL 1194A		Garmin G1000 Software Update 720.17
	2010-2016	SB 1283A		Cabin Altitude Encoder Installation Modification
	2010-2017	SL 1282		Avionics Cooling Fan Advisory
	2010-up	SB 1247B		Garmin SafeTaxi® Limitation
		VSP 212		G1000 Equipped - Momentary Failure Indications
		VSP 216		G1000 Equipped - Momentary Loss of GRS 77/77H and 74A/74B/74H Functions
	2011-2013	VSP 232		Lycoming Low RPM Engine Surge
	2012-2016	SL 1211A		Audio Bus Wiring Inspection and Fuse Replacement
	2012-up	SB 1250		Pilot Operating Handbook Temporary Revision
	2015-2016	SL 1198		Garmin G1000 Software Version 2164.03
		SL 1201B	88589-002	Garmin G1000 Software Version 2164.06 with Stall Heat Control
		VSP 248		Aspen Avionics Evolution Backup Display Software Version 2.6.3 Upgrade
	2015-2017	SL 1250A	88589-002 88642-001	Garmin G1000 Software Version 2164.09
		SL 1253		G1000 Electronic Checklist

CHART 1 (Sheet 5 of 7)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-46-350P	2015-up	VSP 252		Garmin SA 1605 AFCS Armed Mode Cancellation
(Cont.)	2016-2017	SB 1314		Cabin Door Frame Inspection
	2016-2018	SB 1328	46F44A701-701	Aileron Quadrant Cable Guard Replacement
	2017-2019	SB 1343		Canted Frame Doubler Inspection
	2017-up	SL 1246		G1000 NXi Electronic Checklist
		VSP 270		Garmin Loss of ILS Auto Transition
	2018	SB 1336	83287-006	Aileron-Rudder Interconnect Inspection
	2019-2020	SB 1350		Nose Landing Gear Steering Horn and Arm Assemblies Inspection
PA-46R-350T	2008	SB 1190B		Aileron Quadrant Inpsection
		SB 1192	88452-002	Stall Warning Heat Cotnrol Modification
		SB 1193		Nose Gear Down Spring Replacement
		SB 1196		Collector Tank Fuel Drain Inspection
		SB 2000		Current Limiter Inspection
		SL 1111	88439-002	Autopilot Pitch Servo and Programmer
		SL 1113	88444-002	Crew Seat Back Modification
		SL 1114	88446-002	NACA Inlet Guide Vane Installation Kit
		SL 1126A		Voltage Suppressor Advisory
		SL 1170	88542-003	Evaporator Module Assembly Replacement Requirements
	2008-2009	SB 1181B		Flap Track/Splice Strap Inspection/Modification
		SB 1201		Tie Bus Circuit Breaker Inspection
		SB 1210		TWX670 Antenna Inspection
		SB 1212		Inspection of Flap Position Indicator Placard
		SB 1217A	88465-002	Starter Contactor Replacement
		SL 1117		MLB 700 Remote Control Antenna Modification
		SL 1118		Sky-Tec Starter Replacement
		VSP 202		Lycoming SB 590A Use of Aircraft Boost Pump with Kelly Aerospace 200F Series Fuel Pumps
	2008-2010	SB 1208		Lightning Diverter Inspection
		SB 1211		Aft Bulkhead Sealing Modification
		SB 1228	88485-002	Autopilot Circuitry Modification
		SL 1120		Air Conditioning Servicing
		SL 1127		Emergency Exit Interior Panel Hook and Loop Fastener Replacement
		SL1129		Improved Landing Gear Service Replacement Actuators Available
		SL 1136A		Fuel Pressure Switch Adjustment
	2008-2011	SL 1138		Safe Flight Service Letter No. 65, Lift Transducer Modification
		SL 1157		Exterior Lighting - LED Upgrade
		VSP 215		Fuel Drain Valve Adapter Assembly
		VSP 217		Non-Conforming V-Band Clamps
	2008-2012	SB 1243C		Potential Landing Gear Malfunction
		SL 1128		Improved Capacity Battery Replacement
	2008-2013	SB 1258		Fuel Vent Valve Inspection

CHART 1 (Sheet 6 of 7)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-46R-350T	2008-2016	SB 1288A		Aft Vertical Fin Spar Inspection
(Cont.)		SB 1290B		Flap Bracket Inspection, Inboard Bellcrank
		SB 1291	88581-701 88581-702	Executive Table Placards Replacement
		SB 1293C	88584-701 88582-701 88582-702	Engine Mount Doubler Installation and Towing Placards Replacement
		SB 1306	88609-001 88609-002	Wing Rib Modification
		SB 1338		Main Gear Actuator Attach Fitting Inspection
		SB 1327	88630-701	Aileron Control Cable Protection Tube Inspection
		SL 1202		Main Landing Gear Actuator Hydraulic Hose Replacement
		SL 1249		Control Column Plug Modification
	2008-2020	SL 1233		Wing and Flap Protective Tape Improvement
	2008-up	SB 1256		Nose Landing Gear Torque Link Hardware Replacement
		SL 578B		Fuel Line and Support Clamp Inspection and Installation
		SL 1093B		Electrical Contactor Replacement Kit
		SL 1123		Avidyne EXP5000 Software Upgrade (Mod 57)
		SL 1132		Fuel Tank Leaks
		SL 1135		Control Cable / Pulley Inspections
		SL 1159		Fuel Sender Upgrade
		SL 1224		Decorative Window Sealant Improvement
		SSL 437D		Aircraft Tire Options
		VSP 116B		Lycoming SI-1492D Piston Plug Wear Inspection
		VSP 191		Avidyne SA-08-001 Avidyne PFD
		VSP 196		Slick SB2-08A Magneto Inspections Models 4300/6300
		VSP 197A		Slick SB3-08A Magneto Insp. 4200/6200 & 4300/6300
		VSP 207		HET Turbocharger CHRA
		VSP 219		G1000 Equipped - LNAV/VNAV Full Scale Deflection
		VSP 255A		Lycoming AVStar Flow Divider (Lycoming Fuel Manifold) Inspection
		VSP 259		Hartzell Blade - Secondary Windings on ( )7690 and ( )7890 Composite Blades
		VSP 260		Lycoming Identification of Connecting Rods with Non-Conforming Small End Bushings
		VSP 266		Lycoming Recommended Engine Procedures for Purge Vapor During Ground Operations
	2009-2010	SB 1223		Wire Screen Replacement
		SL 1122		G1000 Split Communciations Enable
	2010-2011	SB 1232	100700-014	Cabin Door Handle Placard Inspection
		SL 1140		G1000 Software Version 720.12
	2010-2012	SL 1148		G1000 Software Version 720.14

CHART 1 (Sheet 7 of 7)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-46R-350T (Cont.)	2010-2013	SL 1168		False Alternator Fail Annunciations
		SL 1179		Garmin G1000 Software Ver. 720.15
	2010-2014	SB 1269A		GWX 68 Ground Rewire
		SL 1160		Interior Lighting - LED Upgrade
		SL 1196A		Availability of GTX 33ES Transponder
		SL 1194A		Garmin G1000 Software Update 720.17
	2010-2016	SL 1203A		Garmin G1000 Software 720.17 with Stall Heat Control
		SL 1234		Garmin G1000 GIA Software Update for Transponder Compatability with ADS-B Out
	2010-2017	SL 1282		Avionic Cooling Fan Advisory
	2010-up	SB 1247B		Garmin SafeTaxi® Limitation
		VSP 212		G1000 Equipped - Momentary Failure Indications
		VSP 216		G1000 Equipped - Momentary Loss of GRS 77/77H and 74A/74B/74H Functions

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SPECIAL INSPECTIONS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The following inspections are in addition to those listed in 5-20-00. These inspections are performed at intervals of:

- Flight hours;
- Calendar Year; or
- the specific operation being conducted or the environment being operated in.

Unless otherwise indicated, these inspections are to be repeated at each occurrence of the specified interval. Note that the items listed herein are guidelines based on past operating experience. Each operator should closely monitor his own unique operating conditions/environment and react accordingly to keep his aircraft airworthy.

NOTE: A log book entry should be made upon completion of any inspections.

1. Per Flight Hour

A. First 25 Hours

For a new or newly overhauled engine, after the first 25 hours:

[ ] (1) Drain oil sump while engine is warm.

[ ] (2) Inspect oil lines and fittings for leaks, security, chafing, dents, and cracks.

[ ] (3) Change full-flow (cartridge-type) oil filter element, check element for foreign particles. (See 12-10-00.)

[ ] (4) Fill engine with oil. (See 12-10-00.)

B. First 50 Hours

[ ] Inspect security of injector nozzles.

C. Each 50 Hours

[ ] In PA-46-350P airplanes with pilot's side cockpit windows incorporating storm windows and which have accumulated 350 hours time-in-service, perform the 50 Hour Pilot's Side Cockpit Window/Storm Window Inspection, under Pilot's/Copilot's Window in 56-10-00.

D. Each 250 Hours

[ ] (1) Each 250 hours time-in-service or annually, whichever comes first, comply with the latest revisions of Lycoming Service Bulletins No. 521 and 531.

[ ] (2) Each 250 hours time-in-service or annually, whichever comes first, comply with Piper Service Bulletin No. 1204 dated December 18, 2009. Use the pdf version of the service bulletin found online at http://pubs.piper.com/.

E. Each 400 Hours

[ ] (1) For airplanes equipped with Gill G-243 or G-250S battery, beginning initially at 800 ±50 flight hours or 11 ±1 calendar months, and each 400 ±50 flight hours or 11 ±1 calendar months thereafter, the battery should be removed and capacity tested per Battery Capacity Test, 24-30-00.

[ ] (2) Each 400 hours of engine operation, remove the rocker box covers and check for freedom of valve rockers when valves are closed. Look for evidence of abnormal wear or broken parts in the area of the valve tips, valve keepers, springs, and spring seats. If any indications are found, the cylinder and all of its components must be removed (including the piston and connecting rod assembly) and inspected for further damage. Replace any parts that do not conform with limits shown in the latest revision for Lycoming Service Table of Limits SSP1776.

F. Each 500 Hours

[ ] (1) If installed, replace the vacuum system inlet air filter (i.e., central air filter, gyro filter, etc.) element each 500 hours time-in-service, annually, or at vacuum pump replacement, whichever comes first.

[ ] (2) Replace engine air filter.

[ ] (3) Drain wing fuel tanks.

[ ] (4) Check fuel tank sealing material.

[ ] (5) Check fuel transmitters and electrical connectors for cleanliness, condition, and security.

[ ] (6) Every two (2) years or after 500 hours, whichever occurs first, check that fuel transmitter floats (mounted on forward access panels #2 and #5 from the wing tip) are secure and that fuel strainers (three in each wing tank) are secure and unobstructed. Strainers maybe inspected by removing the forward and aft access covers, fifth in from the wing tip.

[ ] (7) Remove propeller; remove sludge from crankshaft. Clean any residual sludge clinging to the exterior of propeller / crankshaft mating surfaces.

[ ] (8) For airplanes equipped with Slick Magnetos: inspect and clean magneto(s) per 500 Hour Inspection in the Slick F1100 Master Service Manual.

[ ] (9) Coin-tap erosion shield on composite blades. (See latest revision Hartzell Composite Blade Manual No. 135.)

[ ] (10) For airplanes equipped with Garmin G1000/G1000 NXi: each 500 hours time-in-service conduct the 500 Hour Inspection as specified in 34-25-01 (G1000) or 34-25-02 (G1000 NXi), Integrated Avionics System (IAS) - Garmin, Inspections.

[ ] (11) Inspect cabin pressurization system operation. See 21-30-00, Pressurization System Test.

G. Each 600 Hours

[ ] For airplanes equipped with Aero Accessories, Inc. vacuum pump(s) model AA441CC / AA442CW series without Wear Indicator Ports, replace each pump as it accumulates 600 hours time-in-service.

H. First 1000 Hours

[ ] In airplanes equipped with the Garmin GFC 700 autopilot: first 1,000 hours or first three (3) years after installation perform a visual inspection and clean and apply grease to output gear of the GSA 8X Servo Actuator. (See 22-10-00)

I. Each 1000 Hours

[ ] (1) Replace engine compartment flexible fuel and oil hoses as required; but not to exceed 1000 hours time-in-service, eight (8) years, or engine overhaul, whichever comes first; except for TSO-C53a - Type D hoses which are replaced on-condition.

[ ] (2) Replace nose gear hydraulic hoses (gear actuator/door actuator/sequence valve), as required; but not to exceed 1000 hours time-in-service, or eight (8) years, whichever comes first.

[ ] (3) Coin-tap the entire blade on composite blades. (See latest revision Hartzell Composite Blade Manual No. 135.)

[ ] (4) Inspect condition of bolts used with flap rollers and aileron hinges. (Replace as required.)

[ ] (5) Inspect flap tracks for security of attachment, damage, condition, and corrosion.

[ ] (6) If installed, remove SpeedBrake cartridges and return to Precise Flight Inc. for clutch lubrication and spring replacement.

[ ] (7) For airplanes equipped with Garmin G1000/G1000 NXi: each 1000 hours time-in-service conduct the 1000 Hour Inspection as specified in 34-25-01 (G1000) or 34-25-02 (G1000 NXi), IAS, Inspections.

J. Each 2000 Hours

[ ] (1) Each 2000 hours or seven (7) years, whichever occurs first, remove interior panels, and headliner and conduct detailed inspection of aircraft structure (skin, bulkheads, stringers, etc.) for condition and security. Inspection of structure concealed by headliner may be accomplished by alternate means (i.e. - through the use of a borescope) without removing the headliner, providing access is obtained to all concealed areas and borescope provides sufficient detail to adequately accomplish the inspection.

[ ] (2) Each 2000 hours, or as specified in the latest revision of Lycoming Service Instruction No. 1009, overhaul or replace engine.

[ ] (3) Overhaul or replace Hartzell propellers each five or six years or each 2000 or 2400 hours. (Refer to latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

[ ] (4) Each 2,000 Hours or Twelve (12) Years, or at engine overhaul, whichever comes first, overhaul or replace alternators.

K. Each 2,200 Hours

[ ] For airplanes equipped with Aspen Avionics EBD: replace the internal back-up battery in the EBD every three (3) years or 2,200 hours.

L. Each 2400 Hours

[ ] (1) Overhaul or replace Hartzell propeller governors each 2400 hours or at engine overhaul. (Verify TBO in latest revision of Hartzell Service Letter No. 61.)

[ ] (2) Overhaul or replace Hartzell propellers each five or six years or each 2000 or 2400 hours. (Refer to latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

M. Each 2700 Hours

[ ] Overhaul the Sky Tec Starter Model No. 149-NL each 2700 hours or at engine overhaul.

N. Each 4000 Hours

[ ] In PA-46-350P S/N's 4636375 thru 4636462 (less 4636460): each 4,000 hours time-in-service inspect the COMM 1 (lower) Antenna per the COMM 1 (Lower) Antenna Inspection in 23-10-00.

O. Each 5000 Hours

[ ] If installed, remove SpeedBrake cartridges and return to Precise Flight Inc. for drive assembly replacement.

P. Each 7500 Hours

[ ] Replace engine mount bolts located at the firewall (F.S. 79.00).

Q. Each 9400 Hours

[ ] In PA-46-350P S/N's 4636375 thru 4636462, less 4636460: each 9,400 hours time-in-service inspect the COMM 2 (upper) Antenna per the COMM 2 (Upper) Antenna Inspection in 23-10-00.

2. Per Calendar Year

A. Each Thirty (30) Days

NOTE: Pilots/owners may perform and sign-off the following checks:

[ ] (1) If installed, the pilot should check any disposable type (non-gauged) fire extinguisher for condition and charge. Verify nozzle is unobstructed and safety seal is intact. Determine charge by "hefting" extinguisher.

[ ] (2) If installed, the pilot should check gauged rechargeable fire extinguisher for condition and charge. Check to see that the extinguisher is undamaged; that the nozzle is unobstructed; that the gauge pressure is in the operable (green) range; that the lockpin and tamper seal is in place, and that the operating instructions are clearly visible.

B. Each Ninety (90) Days

[ ] (1) Each 90 days or 50 hours, whichever comes first, check fuel filter. Drain and flush completely. Clean or replace as required.

[ ] (2) In PA-46-350P S/N's S/N's 4636375–4636651, less 4636633, check standby attitude indicator emergency power supply (in radar pod aft compartment) for condition and security. Self-test unit using integral test switch - see 34-20-00. Hand-tighten electrical connector.

[ ] (3) In PA-46R-350T only: if the Standby Attitude Indicator has not been operated in the previous 90 days, charge the battery of the Standby Attitude Indicator. See Standby Attitude Indicator, 34-20-00.

[ ] (4) Inspect battery, battery compartment, and vent system for water, corrosion, etching, condition, and security. Flush compartment as required. (Not required if maintenance-free battery installed.)

[ ] (5) Operationally test ELT per the Test procedure in 25-60-00 for the appropriate installed ELT at least every 90 days.

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

[ ] (6) In PA-46R-350T only, if equipped, at least every 90 days, visually check oxygen cylinder installation for leakage, corrosion, bulges, gouges, distortion, and security of mounting.

C. Each Four (4) Months

[ ] Change the engine oil and full-flow cartridge oil filter each four (4) months or every 50 hours time-in-service, whichever comes first.

D. Each Six (6) Months

[ ] (1) If annual usage is significantly less than 100 Hours, lubricate Hartzell propeller each six (6) months. See Hartzell Standard Practices Manual No. 202 latest revision.

[ ] (2) In airplanes equipped with Concorde RG Series batteries and operating less than 1,000 hours per year and battery capacity was at the last capacity check, conduct subsequent checks every six months in service (± 1 month). If battery capacity is less than 85%, remove battery from service.

E. Each Eleven (11) Months

[ ] For airplanes equipped with Gill G-243 or G-250S battery, beginning initially at 800 ±50 flight hours or 11 ±1 calendar months, and each 400 ±50 flight hours or 11 ±1 calendar months thereafter, the battery should be removed and capacity tested per Battery Capacity Test, 24-30-00.

F. First Twelve (12) Months

[ ] In airplanes equipped with Concorde RG Series batteries and operating less than 1,000 hours per year, conduct the initial battery capacity check at 12 months after initial installation (± 1 month).

G. Each Twelve (12) Months

[ ] (1) Lubricate Hartzell propeller every 100 Hours or annually, whichever comes first. If annual usage is significantly less than 100 Hours, lubricate propeller each six (6) months. See Hartzell Standard Practices Manual No. 202 latest revision.

[ ] (2) If installed, each twelve (12) calendar months, conduct SpeedBrake Annual Inspection per Annual Inspection, 27-60-00.

[ ] (3) If installed, replace the vacuum system inlet air filter (i.e., central air filter, gyro filter, etc.) element each 500 hours time-in-service, annually, or at vacuum pump replacement, whichever comes first.

[ ] (4) For vacuum system equipped airplanes, beginning at 5 years from date of check valve manifold manufacture and each 12 months thereafter, inspect the Airborne 1H5 and 1H24 series check valve manifolds per the latest revision of Airborne SL 39A.

[ ] (5) In airplanes equipped with Concorde RG Series batteries and operating less than 1,000 hours per year and battery capacity was above 90% at the last capacity check, conduct subsequent checks every twelve months in service (± 1 month). If battery capacity is less than 85%, remove battery from service.

[ ] (6) Each 250 hours time-in-service or annually, whichever comes first, comply with the latest revisions of Lycoming Service Bulletins No. 521 and 531.

[ ] (7) in PA-46-350P S/N's S/N's 4636375–4636651, less 4636633; and all PA-46R-350T: Each twelve months, perform a full capacity test of the Standby Attitude Indicator battery. See Standby Attitude Indicator, 34-20-00.

[ ] (8) Each 250 hours time-in-service or annually, whichever comes first, comply with Piper Service Bulletin No. 1204 dated December 18, 2009. Use the pdf version of the service bulletin found online at http://pubs.piper.com/.

[ ] (9) For airplanes equipped with Aspen Avionics EBD: each twelve (12) months inspect per the annual inspection under Aspen Avionics Evolution Backup Display (EBD) in 34-24-00.

G. Each Twelve (12) Months (cont.)

[ ] (10) If a rechargeable (gauged) fire extinguisher is installed perform annual maintenance per "Gauged, Class 2B:C Fire Extinguisher Model No. A344T" in 26-20-00.

[ ] (11) Lubricate per annual requirements of Lubrication Chart in 12-20-00.

H. Each Two (2) Years

[ ] (1) Every two (2) years or after 500 hours, whichever occurs first, check that fuel transmitter floats (mounted on forward access panels #3 and #5 from the wing tip) are secure and that fuel strainers (three in each wing tank) are secure and unobstructed. Strainers maybe inspected by removing the forward and aft access covers, fifth in from the wing tip.

[ ] (2) Test and inspect the static pressure system and altimeters. Ensure compliance with the requirements of FAR 43, Appendix E. (See FAR 91.411.) If optional Meggitt EFIS installed, see Appendix 1. If Garmin G1000 or G1000 NXi is installed, see 34-25-01 (Garmin 1000) or 34-25-02 (G1000 NXi), LRU Test Procedures, for GDC 74A (Garmin 1000) or GDC 72 (G1000 NXi) Air Data Computer Tests.

[ ] (3) Test and inspect the transponder. Ensure compliance with the requirements of FAR 43, Appendix F. (See FAR 91.413.)

[ ] (4) Swing the magnetic compass and recalibrate the magnetometer every two (2) years. See Magnetic Heading Systems in 34-20-00.

[ ] (5) For airplanes equipped with Aspen Avionics EBD: each twenty-four (24) months inspect per the twenty-four (24) month inspection in 34-24-00.

[ ] (6) If installed, recalibrate the GAE 43 each twenty-four months per GAE 43 Calibration under LRU Test Procedures in 34-25-01 or 34-25-02, as appropriate.

I. First Three (3) Years

[ ] In airplanes equipped with the Garmin GFC 700 autopilot: first three (3) years or first 1,000 hours after installation perform a visual inspection and clean and apply grease to output gear of the GSA 8X Servo Actuator. (See 22-10-00.)

J. Each Three (3) Years

[ ] (1) In all PA-46R-350T and in PA-46-350P (S/N's 4636460, 4636463–4636632, and 4636634–4636652), replace the Standby Attitude Indicator battery as required, but at least every three (3) years. See Standby Attitude Indicator in 34-20-00.

[ ] (2) For airplanes equipped with Aspen Avionics EBD: replace the internal back-up battery in the EBD every three (3) years or 2,200 hours, or if it fails the operational test.

[ ] (3) In PA-46R-350T airplanes only, overhaul or replace the fixed oxygen system masks each three years or as required. See 35-20-00.

K. Each Five (5) Years

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

[ ] (1) In PA-46R-350T only, remove and hydrostatically test oxygen cylinder (i.e., standard weight cylinders - ICC or DOT “3AA”) every five (5) years or if marked with a five-pointed star, every ten (10) years (or as specified in the latest applicable national regulations).

[ ] (2) Overhaul the regulator. Replacement of the fill valve is recommended, but not required.

[ ] (3) Inspect the Attach Fittings and Push Rods listed below. Repair or replace as required and, when finished, reapply Dinitrol AV8 Corrosion Inhibiting Compound (P/N 89500-800).

[ ] (4) Fwd and aft wing spar to fuselage attach fittings. (See Attach Fittings Corrosion Control, 57-40-00.)

[ ] (5) Fwd vertical fin attach fittings. (See Attach Fittings Corrosion Control, 55-30-00.)

[ ] (6) Elevator trim tab push rods. (See Elevator Trim Tab Push Rod Corrosion Control, 55-20-00.)

[ ] (7) Fwd and aft horizontal stabilizer attach fittings. (See Attach Fittings Corrosion Control, 55-10-00.)

[ ] (8) Overhaul or replace Hartzell propellers each five or six years or each 2000 or 2400 hours. (See latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

[ ] (9) In PA-46-350P S/N’s 4636633, 4636652 and up, the CO Pulse Oximeter/Detector must be returned to CO Guardian at the end of Service Life (Five (5) years) for replacement and calibration of the CO sensor to maintain airworthiness of the unit.

[ ] (10) For airplanes equipped with Garmin G1000/G1000 NXi IAS: update the GRS “Earth magnetic field model” each five (5) calendar years. The updated “Earth magnetic field model” is available from Garmin on July 1st of each fifth year - i.e., 2020, 2025, etc.

L. Each Six (6) Years

[ ] (1) Overhaul or replace Hartzell propellers each five or six years or each 2000 or 2400 hours. (Refer to latest revision of Hartzell Service Letter No. 61 to determine specific requirements for individual airplanes.)

[ ] (2) For airplanes equipped with Aero Accessories, Inc. vacuum pump(s), replace the shear coupling each six (6) years time-in-service.

[ ] (3) If installed, completely discharge the rechargeable (gauged) fire extinguisher and have it professionally inspected and recharged each six years time-in-service.

M. Each Seven (7) Years

[ ] Each 2000 hours or seven (7) years, whichever occurs first, remove interior panels, and headliner and conduct detailed inspection of aircraft structure (skin, bulkheads, stringers, etc.) for condition and security. Inspection of structure concealed by headliner may be accomplished by alternate means (i.e. - through the use of a borescope) without removing the headliner, providing access is obtained to all concealed areas and borescope provides sufficient detail to adequately accomplish the inspection.

N. Each Eight (8) Years

[ ] (1) Replace engine compartment flexible fuel and oil hoses as required; but not to exceed 1000 hours time-in-service, eight (8) years, or engine overhaul, whichever comes first; except for TSO-C53a - Type D hoses which are replaced on-condition.

[ ] (2) Replace nose gear hydraulic hoses (gear actuator/door actuator/sequence valve), as required; but not to exceed 1000 hours time-in-service, or eight (8) years, whichever comes first.

O. Each Ten (10) Years

[ ] (1) Each ten (10) years time-in-service, test fuselage and wing fluid hoses to system pressure. Visually inspect for leaks. Hoses that pass inspection may remain in service, but must be rechecked each five (5) years additional time-in-service. No fluid hose may exceed twenty (20) years total time-in-service.

[ ] (2) For airplanes equipped with Avidyne Entegra: replace the CMOS battery in the multifunction display (MFD) as required, but at least each 10 years. See Multifunction Display in 34-22-00.

[ ] (3) For vacuum system equipped airplanes, replace Airborne 1H5 and 1H24 series check valve manifolds at 10 years from check valve manifold date of manufacture. See latest revision of Airborne SL 39A.

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

[ ] (4) In PA-46R-350T only, for standard weight oxygen cylinders (ICC or DOT “3AA”) marked with a five-pointed star, remove and hydrostatically test every ten (10) years (or as specified in the latest applicable national regulations).

[ ] (5) In PA-46-350P only, replace Fixed Oxygen Generators at ten (10) years from date of manufacture as indicated on the unit data plate.

P. Each Twelve (12) Years

[ ] (1) If installed, replace disposable-type (non-gauged) fire extinguishers at twelve (12) years from date of manufacture.

[ ] (2) If installed, hydrostatically test rechargeable (gauged) fire extinguishers at twelve (12) years time-in-service.

[ ] (3) Each 2,000 Hours or Twelve (12) Years, or at engine overhaul, whichever comes first, overhaul or replace alternators.

Q. Each Fifteen (15) Years

[ ] In PA-46R-350T only, replace DOT-SP-10945 oxygen cylinder at fifteen (15) years from date of manufacture.

R. Each Twenty (20) Years

[ ] No fluid hose may exceed 20 years total time-in-service.

3. Per Specific Operation / Operating Environment

A. Operation in High Dust or Industrial Pollution Environment

CAUTION: DISCONNECT LINES FROM PITOT/STATIC SYSTEM BEFORE CONDUCTING THIS INSPECTION.

Item	Inspection	Inspection Interval
☐ Engine Air Filter.	Clean and inspect.	Daily.
☐ Cabin Environmental and Instrument Air Filters.	Inspect and replace if necessary.	100 Hours.
☐ Pitot/Static system.	Check for obstruction. Reverse flow to lines.	100 Hours or as required.
☐ Landing Gear Actuators and Oleos.	Clean.	Before each flight.
	Inspect.	100 Hours.
☐ Landing Gear Wheel Bearings.	Clean, inspect and repack.	50 Hours.
☐ Windows.	Inspect for cracks, erosion, crazing, visibility, and cleanliness.	Daily.
☐ Structure drain holes.	Clean with pipe cleaner.	Before each flight.

B. Operation in High Salt or High Humidity Environment

Item	Inspection	Inspection Interval
☐ Fuselage, Empennage and Wings.	Remove floor panels and exterior access plates; inspect for corrosion.	200 Hours.
☐ Landing Gear.	Inspect for corrosion and lubrication; switches and wiring for condition.	200 Hours.

WARNING: ENSURE BOTH MAGNETO SWITCHES ARE OFF (GROUNDED), BEFORE TURNING PROPELLER. ENGINE MAY START IF BOTH SWITCHES ARE NOT OFF. USE EXTREME CAUTION WHEN ROTATING PROPELLER BY HAND; PROPELLER MAY KICK BACK.

Item	Inspection	Inspection Interval
☐ Engines with more than 50 hours total time.	Each five days, pull prop through five complete revolutions. Each 30 days, fly aircraft for 30 minutes or, ground run until oil temperature is in the green arc. Avoid excessive ground run.	Each 5 days and each 30 days.
☐ Engines with less than 50 hours total time.	Each day, pull prop through five complete revolutions. Each 30 days, fly aircraft for 30 minutes or, ground run until oil temperature is in the green arc. Avoid excessive ground run.	Daily and each 30 days.
☐ Instruments and Wiring.	Inspect for proper seal of cases and corrosion.	100 Hours.
☐ Interior.	Inspect upholstery, seat belts, seats and rugs for corrosion and integrity.	100 Hours.

NOTE: Do not use metallic tie downs (i.e. - chains, cables, etc.) in high salt or high humidity environments.

C. Operation in Extreme Cold

	Item	Inspection	Inspection Interval
☐	Hydraulic, Pneumatic and Environmental.	Check all fittings and attachments for security and leaks.	First 100 Hour, then as required.

D. Operation from Soft or Unusual Terrain

	Item	Inspection	Inspection Interval
☐	Landing Gear.	Inspect for cracks, attachment, damage, cleanliness and lubrication.	100 Hours.
☐	Wheels.	Inspect for cracks, damage, chipped rims; bearings for damage, corrosion and lubrication.	100 Hours.
☐	Tires.	Inspect for cuts, wear, inflation and deterioration.	Daily.
☐	Wheel Wells.	Inspect for foreign material, damage and corrosion.	100 Hours.
☐	Brakes.	Inspect for damage, foreign material, cracks and overheating.	Daily.
☐	Flaps, Lower Fuselage and Wing.	Inspect for damage, cracks and corrosion.	100 Hours.

E. In PA-46R-350T S/N's 4692001 thru 4692135

For airplanes which have not yet complied with Part II of Piper Service Bulletin No. 1211:

(1) **Prior to Each Flight**

- ☐ Conduct Drain Hole Inspection, 53-20-00.

(2) **Immediately Following Flight into Rain**

- ☐ Conduct Drain Hole Inspection, 53-20-00.

(3) **Immediately Following Airplane Wash**

- ☐ Conduct Drain Hole Inspection, 53-20-00.

(4) **For Airplanes Stored Uncovered, within one (1) week of each exposure to rainy environmental conditions**

- ☐ Conduct Drain Hole Inspection, 53-20-00.

F. At Engine Overhaul

	Item	Inspection	Inspection Interval
☐	Oil Cooler.	Replace or overhaul at engine overhaul.	Each Occurrence.

G. Before Each Flight

	Item	Inspection	Inspection Interval
☐	H3R Fire Extinguishers, Model RT-A600, with date codes 2012 and later.	Weigh unit. Weight must be above the minimum specified on the nameplate.	Each Flight.

NOTE: See 26-20-00.

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UNSCHEDULED MAINTENANCE CHECKS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The following inspections are required in response to specific anomalies encountered during aircraft operation. Note that the items listed herein are guidelines based on past operating experience. Each operator should closely monitor his own unique operating conditions/environment and react accordingly to keep his aircraft airworthy.

NOTE: A log book entry should be made upon completion of any inspections.

1. Lightning Strike

	Item	Inspection	Inspection Interval
☐	Propeller.	Hartzell Propellers - refer to the inspection requirements in the latest revision of the Propeller Owner's Manual No. 115N.	Each occurrence, before further flight.
☐	Propeller Deice System. (Opt. 3-blade only.)	Replace MOV module(s).	Each occurrence, before further flight.
☐	Engine.	See latest revisions of appropriate Lycoming Service Bulletins and Overhaul Manual.	Each occurrence, before further flight.
☐	Electrical and Avionics Systems.	Inspect and check harness, connections, and equipment for high voltage damage, burns and insulation degradation. Replace or overhaul as required. Consult with appropriate avionics vendor(s) for inspections and operational checks. Bench check alternator and voltage regulator(s), see 24-30-00.	Each occurrence, before further flight.
☐	Antennas	Inspect all antenna connections for condition and security.	Each occurrence, before further flight.
☐	All exterior surfaces, skins, and structure.	Inspect for burns, evidence of arcing, and damage on surfaces and bearings. Check for correct material properties in the area of the strike path. Degauss engine mount. Replace or repair affected areas/parts.	Each occurrence, before further flight.
☐	System Components.	Inspect instrumentation, hydraulic, vacuum, pitot/static, and fuel systems, for damage and correct operation.	Each occurrence, before further flight.
☐	Static Wicks.	Replace.	Each occurrence, before further flight.
☐	Bearings.	Inspect all control surface hinges and bearings, and landing gear and wheel bearings for pitting and damage. Replace as required.	Each occurrence, before further flight.

2. Engine Overspeed, Overtemp, Loss of Oil, or Sudden Stoppage

	Item	Inspection	Inspection Interval
☐	Engine.	See latest revision of Lycoming Service Bulletin 369J and Overhaul Manual.	Each occurrence, before further flight.
☐	Propeller.	Hartzell Propellers - refer to the inspection requirements in the latest revision of the Propeller Owner's Manual No. 115 and Hartzell Service Letter 61-251.	Each occurrence, before further flight.
☐	Engine Mount and Attachments.	Inspect for distortion and damage. Replace or repair as required.	Each occurrence, before further flight.

3. Severe Turbulence, Hard or Overweight Landing

CAUTION: MINOR OR APPARENTLY SUPERFICIAL DAMAGE MAY INDICATE A MORE SEVERE CONDITION SOMEWHERE ELSE IN THE STRUCTURE.

A. Place aircraft in a normal level attitude.

B. Make a preliminary inspection of checking alignment and out-of-track condition of engine, wings, tail, landing gear and doors.

C. Follow Piper and Lycoming Maintenance Manual procedures. If there are any questions regarding repairs or procedures, contact your Piper Distributor's Service Advisor (DSA).

D. Inspect the following items closely to determine the extent of damage:

	Item	Inspection	Inspection Interval
☐	Landing Gear Struts. (Not required for severe turbulence.)	Cracks, signs of overstress deformation, loose or damaged trunnion mounts. Axles for cracks, bending or flat spots. Damaged oleos and seals, hydraulic leaks and landing gear alignment.	Each occurrence, before further flight.
☐	Wheels, Tires, Brakes. (Not required for severe turbulence.)	Cracks, chips, loose or cracked mounting bolts, alignment of slippage marks, sidewall distress, hydraulic or air leaks. Inspect the wheels (dye penetrant method) and wheel bolts (magnetic particle method).	Each occurrence, before further flight.
☐	Wheel Wells and Landing Gear attach points. (Not required for severe turbulence.)	Buckling, cracks, overstress, wing skin buckling, actuator and side brace damage and condition. Magnaflux landing, gear attachment and drag link bolts.	Each occurrence, before further flight.

3. Severe Turbulence, Hard or Overweight Landing (continued)

Item	Inspection	Inspection Interval
☐ Wings.	Wing attach bolts for slippage, damage and overstress. Fasteners securing wing attachment fittings to fuselage frames and wing spars for condition and security. Upper and lower wing skins for wrinkles, cracks, popped or loose rivets.	Each occurrence, before further flight.
☐	Remove access plates and inspect for internal damage to ribs, stringers and sparwebs; and fuel tanks for damage, attachment, and leaks.	
☐ Engine.	Engine mounts for distortion and damage to elastomeric parts and 100 Hour Inspection in 71-20-00. Propeller for evidence of ground strike (i.e. - hard or overweight landing).	Each occurrence, before further flight.
☐ Fuselage.	Loose or missing rivets, door alignment, windows and attachments for overstress, cracks or damage. Wing carry through member for overstress damage. Stringers, bulkheads, keel beams for buckling, cracks, or damage. Forward and aft pressure bulkheads for buckling, cracks and damage. Avionics, instruments and accessories installation for security and operation.	Each occurrence, before further flight.
☐	Perform ground pressurization check. (See 21-30-00 - Pressurization System Test.)	
☐ Empennage.	Skins for buckling wrinkles, loose or missing rivets. Elevator, rudder, vertical fins and horizontal stabilizer for security of attachment and overstress of bolts. Ribs, stringers for buckling, cracks and damage.	Each occurrence, before further flight.
☐ Wheels, tires, brakes. (Not required for severe turbulence)	Cracks, chips, loose or cracked mounting bolts, alignment of slippage marks, sidewall distress, hydraulic or air leaks. Inspect the wheels (dye penetrant method) and wheel bolts (magnetic particle method.)	Each occurrence, before further flight

4. Flaps Extended Above Maximum Flap Extension Speed (V_{FE})

Item	Inspection	Inspection Interval
☐ Flap bellcranks.	Inspect for distortion. Replace as required. (See Flap Bellcrank Distortion Inspection, 27-50-00.)	Each occurrence, before further flight.
☐ Flaps.	Inspect for damage to the skin and attach points. Replace as required.	Each occurrence, before further flight.

5. Flood Damage, Immersion in Water

A. These guidelines are general in nature and should be applied or varied to fit the individual aircraft according to water level, length of time of exposure and other variables. Only those areas that might not be obvious to the mechanic are addressed.

CAUTION: MAKE ALL REPAIRS AND/OR ADJUSTMENTS IN ACCORDANCE WITH THE APPROPRIATE PIPER MAINTENANCE MANUAL, THE COMPONENT MANUFACTURER'S MAINTENANCE MANUAL, AND FAR PART 43. PAY PARTICULAR ATTENTION TO SILT, CORROSION AND CONTAMINANTS.

B. Follow Piper and Lycoming Maintenance Manual procedures. If there are any questions regarding repairs or procedures, contact your Piper Distributor's Service Advisor (DSA).

C. Determine the water level on the aircraft. Determine which operating and/or electrical components have been exposed to the water.

D. If the following items were immersed, inspect them closely to determine the extent of damage:

Item	Inspection	Inspection Interval
☐ Airframe	Clean silt and contaminants from airframe.	If immersed, each event, before further flight.
☐ Tubular Structures (i.e. - Engine Mounts, etc.)	Check for internal corrosion. Clean and represerve as required. (See 71-20-00 - Engine Mount Corrosion Inspection, Immersion in Water.)	If immersed, each event, before further flight.
☐ Wings	Inspect to ensure that contaminants are cleaned from fuel cell areas.	If immersed, each event, before further flight.
☐ Landing Gear and associated Bearings, Locks, Torque Links, Shimmy Dampeners, etc.	Check all limit switches, replace non-sealed type. Jack airplane and cycle landing gear to ensure proper operation.	If immersed, each event, before further flight.
☐ Control Surfaces	Remove surface, clean and check all bearings - relube or replace as necessary. Rebalance before installation.	If immersed, each event, before further flight.
☐ Flight Control System	Clean and inspect all cables, pulleys, and bearings for evidence of corrosion. Replace corroded cables. Re-preserve galvanized cable with MIL-C-11796 Class 2 (hot).	If immersed, each event, before further flight.

5. Flood Damage, Immersion in Water (continued)

Item	Inspection	Inspection Interval
☐ Trim Control System	Clean and inspect all trim system cables, pulleys, drums, bearings, jack screws, etc. Do not apply preservation to trim cables.	If immersed, each event, before further flight.
☐ Actuating Cables	Inspect “push-pull” actuating cables for powerplant, heating and ventilating system, fuel system, etc. for proper operation.	If immersed, each event, before further flight.
☐ Engine	Remove, disassemble, and inspect. Examine all parts paying particular attention for evidence of corrosion, rust or contaminants imbedded on bearing surfaces, piston, mounting flanges or any aluminum, magnesium or bronze surface that may be porous.	If immersed, each event, before further flight.
☐	Remove evidence of rust, or corrosion. If pitting in stressed areas is found the part should not be reused. Silt imbedded in porous surfaces may be removed. Be certain oil passages, dowel holes and similar hidden openings and recesses are thoroughly free from contaminants.	
☐	Test electrical components and fuel metering devices in accordance with manufacturer’s instructions to determine fitness for future use.	
☐	Reassemble engine using new seals, gaskets, stressed bolts nuts and crankshaft sludge tubes. All reused parts must conform with Lycoming Table of Limits No. SSP-1776 for fits and clearances.	
☐	See latest revision of Lycoming Service Bulletin No. 357.	
☐ Engine Accessories	Inspect. Aircraft systems that supply either fuel or oil to the engine must be thoroughly cleaned, including oil cooler, lines,valves, etc. to prevent contamination of the engine after reinstallation.	If immersed, each event, before further flight.
☐ Propellers	Inspect and repair as necessary in an authorized propeller shop.	If immersed, each event, before further flight.
☐ Hydraulic System	Replace hydraulic powerpak.	If immersed, each event, before further flight.

5. Flood Damage, Immersion in Water (continued)

Item	Inspection	Inspection Interval
☐ Electrical Systems	Replace all circuit breakers and switches (except sealed landing gear limit switches).	If immersed, each event, before further flight.
☐	Replace all solenoids, relays and master contactors.	
☐	Replace battery.	
☐	Disassemble all connectors; clean and inspect for corrosion. Replace all corroded or pitted connectors. Inspect for wire corrosion at connector.	If immersed, each event, before further flight.
☐	Check all harness assemblies for entrapped contaminants. Clean and check for short circuits.	
☐	Remove electric motors and electric pumps.	
☐	Remove all potted solid state electrical equipment such as flap time delay relays, alternator inop. switches, windshield heat timers, low fuel warning switches, etc. Clean, dry and bench check per appropriate maintenance manual.	
☐	Replace de-icer timers. Clean and check prop de-icer brush holders.	
☐	Clean and check voltage regulators and overvoltage relays. Replace as necessary	
☐	Clean and check all strobe light power supplies. Refer to appropriate maintenance manual.	
☐	Replace all fuel senders, trim tab sender pots, flap position senders, etc.	
☐	Clean, inspect and check heated pitot systems.	
☐	Inspect and check harness, connections, and equipment for high voltage damage, burns and insulation degradation. Replace or overhaul as required. Consult with appropriate avionics vendor(s) for inspection and operational checks. Bench check alternator and voltage regulator(s) (See 24-30-00).	Each occurrence, before further flight.
☐ Autopilot System	Bench check in accordance with appropriate maintenance manual. Pay particular attention to clutch settings.	If immersed, each event, before further flight.

5. Flood Damage, Immersion in Water (continued)

Item	Inspection	Inspection Interval
☐ Vacuum and Pitot-Static Systems	Replace gyros.	If immersed, each event, before further flight.
	Replace filters.	
☐	Clean and inspect all lines, and pitot and static vents.	
☐	Clean and check all regulating valves.	
☐	Remove and inspect engine driven and auxiliary vacuum pumps.	
☐ Induction and Bleed Air Systems	Clean and inspect for silt and corrosion. Check all ducts and gaskets. Replace as necessary.	If immersed, each event, before further flight.
☐	Clean and inspect all heat shrouds and ducting.	
☐ Fuel Systems	Remove access panels and clean fuel tank (wet wing) and clean all associated lines and pumps and header tanks.	If immersed, each event, before further flight.
☐	Clean and inspect all fuel tank vents, cap vents and vent lines.	
☐ Instruments	Clean and inspect instruments. Bench check per appropriate maintenance manual.	If immersed, each event, before further flight.
☐ Heating and Ventilating Systems	Replace blowers.	If immersed, each event, before further flight.
	Clean and inspect all distribution boxes, ducting and valves.	
☐	Inspect and check system control cables. Replace corroded or binding cables.	
☐	Clean and inspect air conditioning evaporator, condenser, and compressor.	
☐ Pressurization Systems (PA-46-350P only.)	Remove and replace all system controls and valves.	If immersed, each event, before further flight.
☐ Oxygen Systems	Disconnect all lines from source and outlets; clean all fittings and lines per MIL-I-5585A.	If immersed, each event, before further flight.
☐	Remove and clean regulator valve per appropriate Scott publication.	
☐	Replace pressure gauge.	
☐	Replace oxygen generator units.	

5. Flood Damage, Immersion in Water (continued)

Item	Inspection	Inspection Interval
☐ Avionics Systems	Replace avionics. Clean and inspect antennas and connectors.	If immersed, each event, before further flight.
☐	Inspect and check harness, connections, and equipment for high voltage damage, burns and insulation degradation. Replace or overhaul as required. Consult with appropriate avionics vendor(s) for inspection and operational checks. Bench check alternator and voltage regulator(s) (See 24-30-00).	Each occurrence, before further flight.
☐ Insulation and Upholstery	Remove all wet insulation and upholstery. Thoroughly clean and dry (or replace) to ensure corrosion is not promoted in adjacent structures.	If immersed, each event, before further flight.

CHAPTER

6

DIMENSIONS AND AREAS

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CHAPTER 6

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
6-List of Effective Pages	1	Mar 25/21			
	2	Sep 15/09			
6-Table of Contents	1	Mar 25/21			
	2	Sep 15/09			
6-00-00	1	Mar 25/21			
	2	Mar 25/21			
	3	Mar 25/21			
	4	Mar 25/21			
6-10-00	1	Mar 25/21			
	2	Mar 25/21			
6-20-00	1	Mar 25/21			
	2	Mar 25/21			
6-40-00	1	Mar 25/21			
	2	Mar 25/21			
	3	Mar 25/21			
	4	Mar 25/21			

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CHAPTER 6 - DIMENSIONS AND AREAS

TABLE OF CONTENTS

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GENERAL

Leading Particulars

See "Chart 1".

CHART 1 (Sheet 1 of 3)
LEADING PARTICULARS

MODEL	**PA-46-350P / PA-46R-350T**	
ENGINE		
Manufacturer	Lycoming	
Model	TIO-540-AE2A	
FAA Type Certificate	E14EA	
Rated Horsepower (Sea Level)	350	
Rated RPM	2,500	
Oil Specification	Refer to 12-10-00 and latest revision of Lycoming Service Bulletin "Fuel and Oil Grades".	
Oil Sump Capacity	12.0 quarts	
Fuel Specification	100 or 100LL (Aviation Grade)	
Fuel Injection System	Lycoming	
Magnetos (Pressurized)	Slick	
Left	6363	
Right	6360	
Magneto Timing	20° BTC	
Magneto Breaker Point Clearance	.008 - .012 (Using "E"-gap method)	
Spark Plugs (Shielded):	Refer to latest revision of Lycoming SB No. 1042	
Spark Plug Gap Setting	.019	
Firing Order	1, 4, 5, 2, 3, 6	
Starter (24 volt)		
Hartzell Engine Technologies (HET) (S/N's 4636001–46360402)	MHB 4016 or MHB 6016	
Sky Tec (S/N's 4636403 and up / 4692001 and up)	149NL	
Alternator - (Two) Belt Driven (24 volt)	HET #	Lycoming #
HET (70 amp) (S/N's 4636001–4636131) (1)	ALU-8521-LS	LW-14326
HET (75 amp) (S/N's 4636132–4636644 / 4692001–4692212) (1)	ALU-6539-LS	32C22515
	HET #	Piper #
HET (75 amp) Right (S/N's 4636645 and up / 4692213 and up)	ALU-6539-3	680-515
HET (75 amp) Left (S/N's 4636645 and up / 4692213 and up)	ALU-6539-4	680-516
Voltage Regulator (Two)		
Lamar Inc.	B-00382-1	

(1) Obtain from Lycoming.

CHART 1 (Sheet 2 of 3)
LEADING PARTICULARS

MODEL	PA-46-350P / PA-46R-350T	
PROPELLER		
Manufacturer	Hartzell	
Type	Constant Speed	
Hub	HC-I2YR-1BF (1)	HC-I3YR-1E (2) (3)
	2-Blade	**3-Blade**
Blade	F8074 (4)	7890B (6)
	F8074K (5)	7890K (7)
Diameter (Inches)		
Minimum	79	80
Maximum	80	80
Blade Angle, Low Pitch (High RPM)	17.6° ± 0.2°	13.65° ± 0.15°
Blade Angle, High Pitch (Low RPM)	40.0° to 41.0°	38.7° ± 0.5°
Manufacturer	Hartzell	
Type	Constant Speed	
Hub	HC-I3YR-1N (8)	HC-I3Y1R-1N (8)
	3-Blade	
Blade	N7605 +2 (6)	N7605 +2 (6)
	N7605K +2 (7)	N7605K +2 (7)
		N7605C +2 (6)
		N7605CK +2 (7)
Diameter (Inches)		
Minimum	80	
Maximum	80	
Blade Angle, Low Pitch (High RPM)	14.0° ± 0.2°	
Blade Angle, High Pitch (Low RPM)	38.0° ± 1.0°	
Governor Model (Hartzell)	V-5-2 (9)	
	V-11-1 (10) (11)	
	S-1-30 (8)	

(1) Standard in S/N's 4636001 thru 4636374.
(2) Optional in S/N's 4636132 thru 4636374. Standard in S/N's 4636375 and up.
(3) S/N's 4692001 and up.
(4) Aluminum without de-ice option.
(5) Aluminum with de-ice option.
(6) Composite without de-ice option.
(7) Composite with de-ice option.
(8) Circa 2009 and up.
(9) Original equipment in S/N's 4636001 thru 4636158.
(10) S/N's 4636159 up to circa 2009 and service replacement in S/N's 4636001 thru 4636158.
(11) S/N's 4692001 up to circa 2009.

CHART 1 (Sheet 3 of 3)
LEADING PARTICULARS

MODEL	PA-46-350P / PA-46R-350T
FUEL SYSTEM	
Fuel Tank Capacity (Gal.)	61 per wing
Total Fuel Onboard (Gal.)	122
Total Usable Fuel (Gal.)	120
Fuel	
Minimum Grade	100 / 100LL (Aviation Grade)
LANDING GEAR	
Type	Retractable - Hydraulic
Shock Strut Design	Combination Air and Oil
Turning Radius	35.4 ft
Nose Gear Travel	30° Left and Right
Wheel, Nose	McCauley D-30500, Cleveland 40-77B
Wheel, Main	Parker Hannifin 40-120C
Brakes	Cleveland 30-83A
Tire, Main	6.00 x 6, 8 ply rating
Tire, Nose	5.00 x 5, 6 ply rating
Tire Pressures (12)	(PIR-PPS50025, Rev. AK.)
Main	
Unloaded (on jacks)	55 psi max.
Loaded (weight on wheels)	55–57 psi
Nose	
Unloaded (on jacks)	50 psi max.
Loaded (weight on wheels)	50–52 psi

(12) Pressures specified apply when tires are at ambient temperature.

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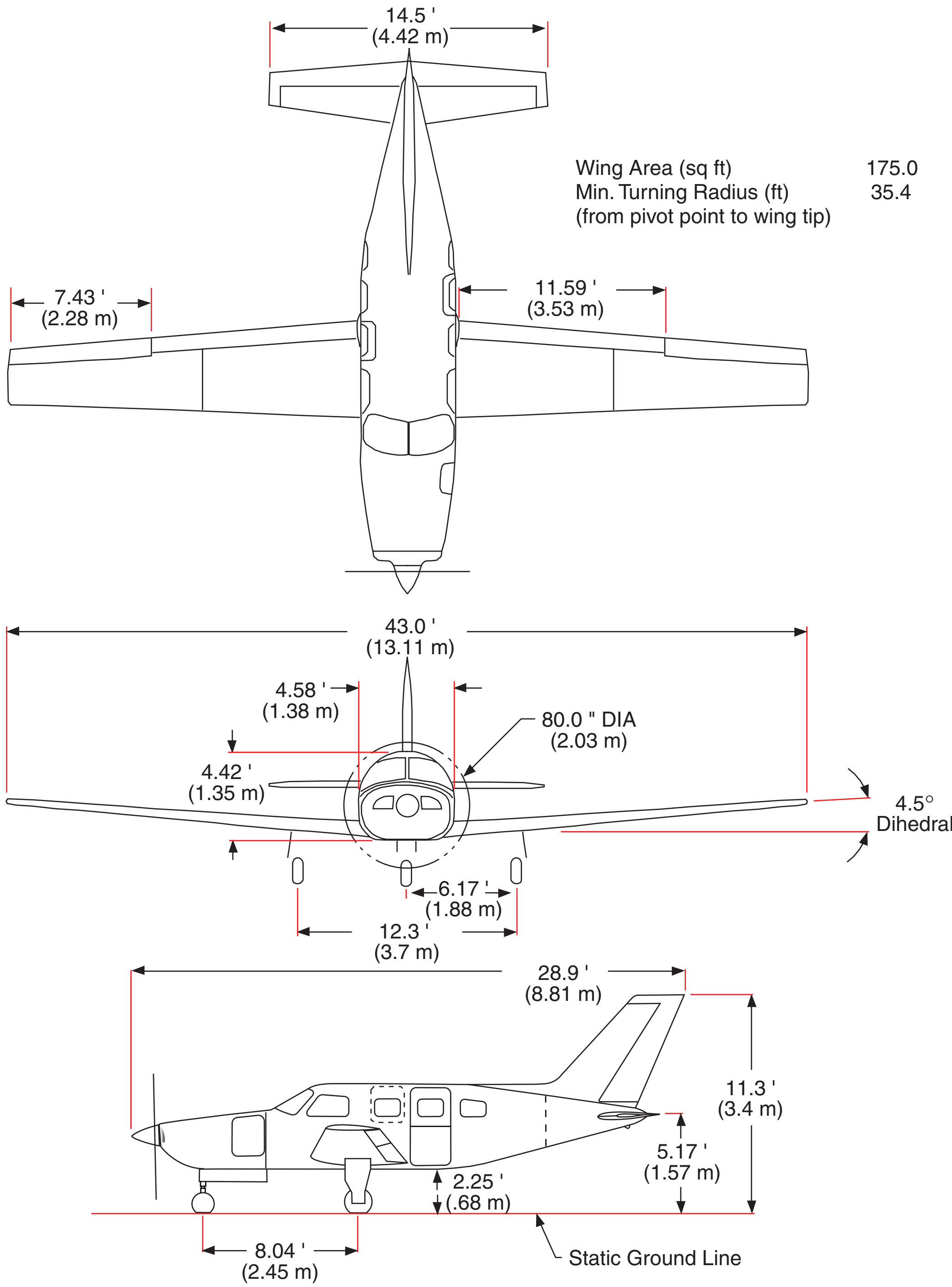
PRINCIPAL DIMENSIONS

The principal airplane dimensions are shown in “Figure 1”. The airplane serial number is located on the Manufacturers Aircraft Association (MAA) plate, which is located below the tailcone at approximately F.S. 311.50. The engine serial number plate is located on the right side of the engine between cylinders 1 and 3.



Three View (PA-46-350P / PA-46R-350T)
Figure 1

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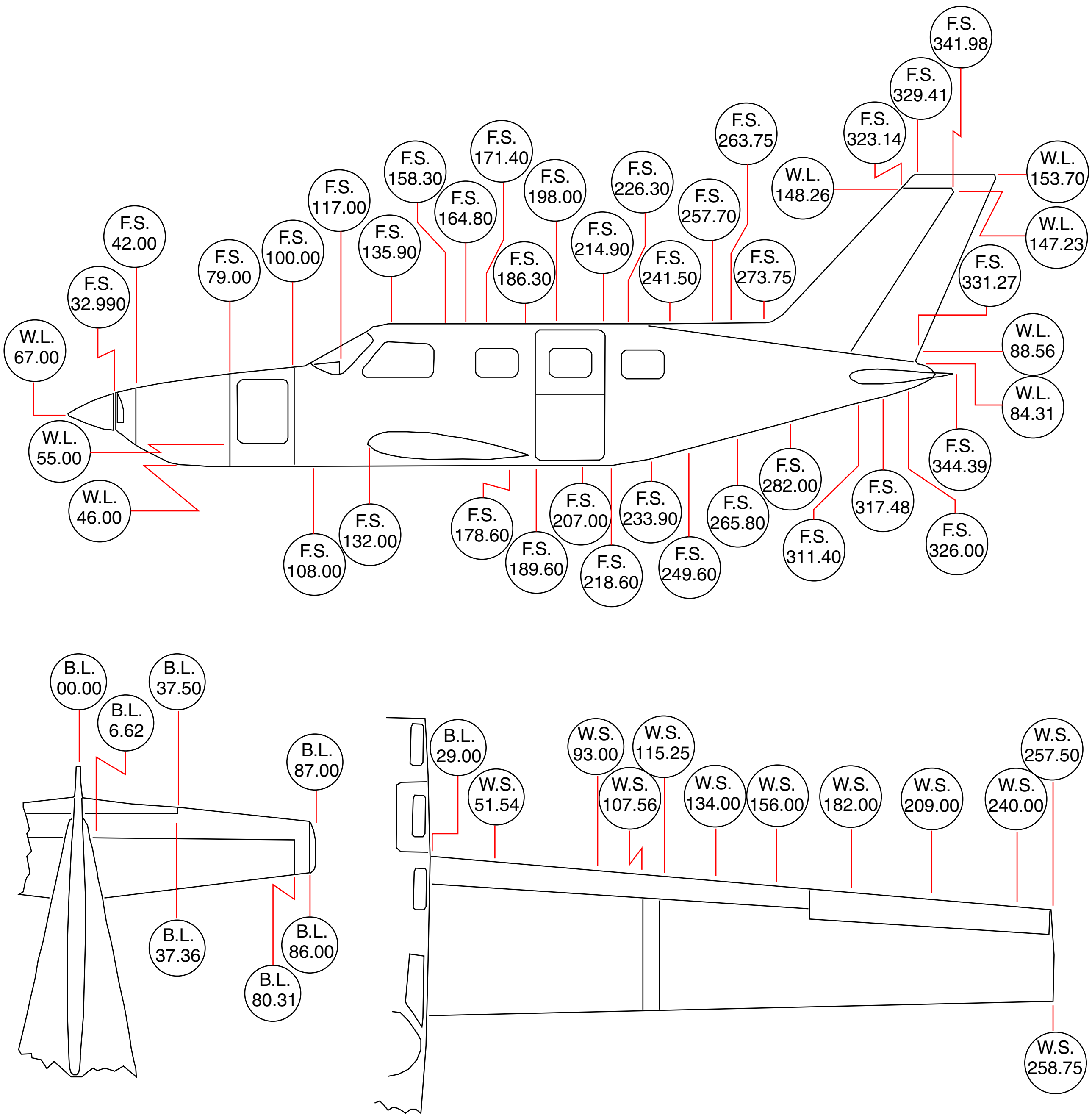
REFERENCE LINES

Station References

To locate various airplane components that require maintenance and servicing, a method using wing fuselage station, wing station, buttock line and waterline designations is frequently used in this manual. Fuselage stations (F.S.), buttock lines (B.L.), and waterlines (W. L.) are reference points measured by inches in the vertical or horizontal direction from a given reference line. These points indicate station locations of airplane structural members.



Station References
Figure 1

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ACCESS PROVISIONS

Access and Inspection Provisions

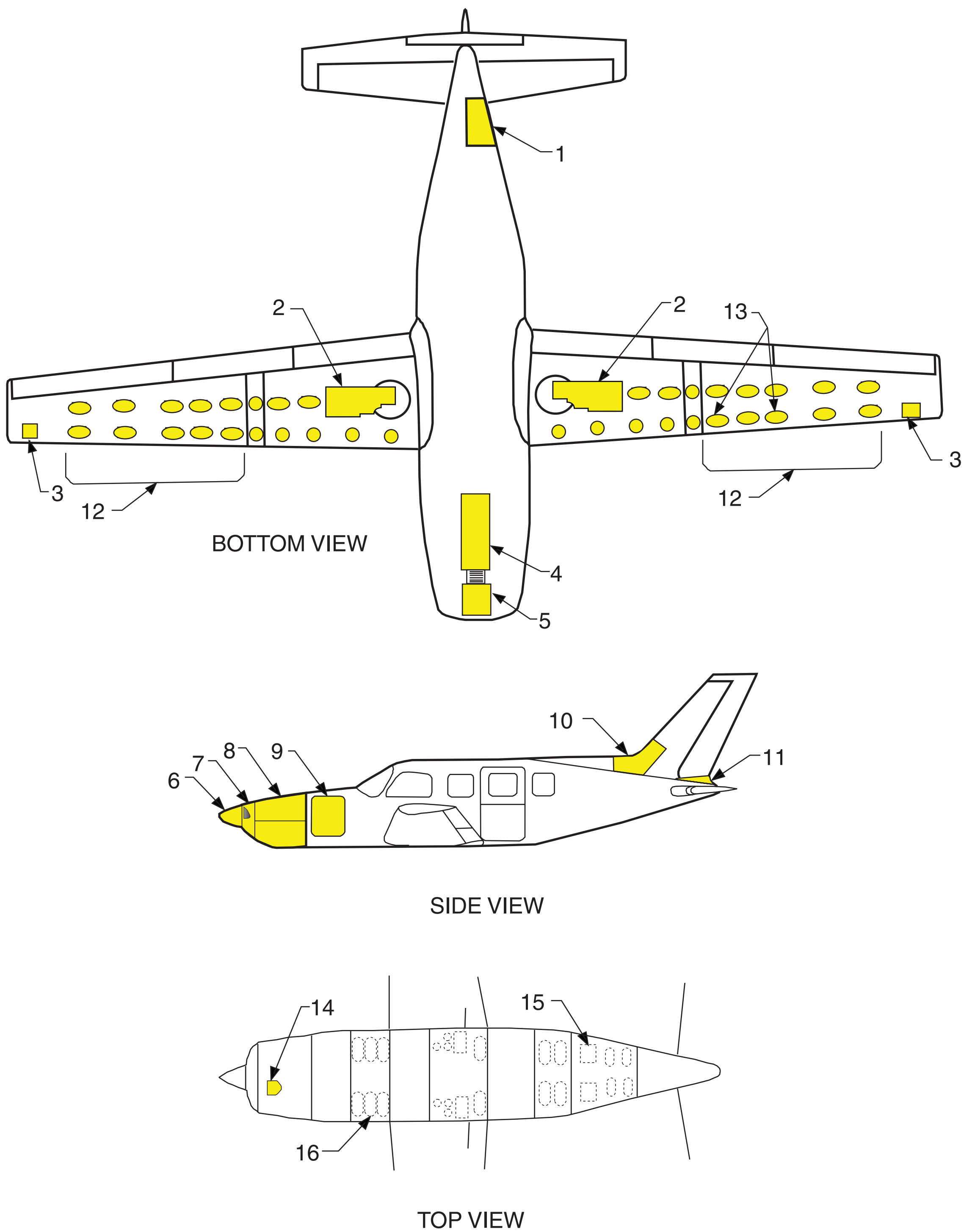
Airplane access and inspection provisions are shown in “Figure 1”. Components to be serviced or inspected through each opening are identified in the illustration by an assigned index reference number. All access plates and panels are secured by metal fasteners or screws.

“Access plates and panels” includes all forms of covers, cowlings, fairings, plates, panels, tips, etc. identified in “Figure 1” which may be removed for access to any component or space.

NOTE: Review Consumable Materials, 91-10-00, for sealant requirements before removing any access plates or panels.

Gain access to floor panel attachment screws by removing seats and carpeting. To enter aft section of the fuselage, remove access panels on the lower side of the fuselage, aft of station 265.00.

All access plates and panels should be inspected for damage or excessive wear, including all mounting holes, each time they are removed. If damage or excessive wear is detected, installation of a new access plate or panel is recommended.



Access Plates and Panels
Figure 1(Sheet 1 of 3)

LEGEND (BY ACCESS PANEL index number)

Index No.	**Components**
1.	Elevator Trim Jackscrew Elevator Up Stop Elevator Sector Rudder Sector Rudder Stops Emergency Locator Transmitter Elevator Cable Turnbuckle A/C Condenser and Fan
2.	Collector Tank Main Landing Gear Actuator Flap Bellcranks Up Limit Switch Squat Switch (L/H side)
3.	Strobe Power Supply PA-46-350P (S/N's 4636001 thru 4636626) PA-46-350T (S/N's 4692001 thru 4692208)
4.	Nose Landing Gear Actuator Steering Bellcrank Steering Bungee
5.	Induction Air Filter
6.	Propeller Attaching Bolts
7.	Alternator(s) Deicer Brush Block
8.	Engine Oil Filter Fuel Filter
9.	Battery, External Power Supply Plug Brake Reservoir Vacuum Regulator(s)
10.	ELT Antenna
11.	Elevator Trim Tab Adjustment
12.	Fuel Tank Access Cover (Wet Wing)
13.	Fuel Transmitters
14.	Oil Dipstick
15.	Hydraulic Pump/Reservoir
16.	Annunciator Dimmer Assembly

Access Plates and Panels
Figure 1 (Sheet 2 of 3)

LEGEND (BY COMPONENT name)

Component	Index No.
A/C Compressor	1
Alternator(s)	7
Annunciator Dimmer Assembly	16
Battery, External Power Supply Plug	9
Brake Reservoir	9
Collector Tank	2
Deicer Brush Block	7
Elevator Cable Turnbuckle	1
Elevator Sector	1
Elevator Trim Jackscrew	1
Elevator Trim Tab Adjustment	11
Elevator Up Stop	1
ELT Antenna	10
Emergency Locator Transmitter	1
Engine	8
Flap Bellcranks	2
Fuel Filter	8
Fuel Tank Access Cover (Wet Wing)	12
Fuel Transmitters	13
Hydraulic Pump/Reservoir	15
Induction Air Filter	5
Main Landing Gear Actuator	2
Nose Landing Gear Actuator	4
Oil Dipstick	14
Oil Filter	8
Propeller Attaching Bolts	6
Rudder Sector	1
Rudder Stops	1
Steering Bellcrank	4
Steering Bungee	4
Strobe Power Supply PA-46-350P (S/N's 4636001 thru 4636626) PA-46-350T (S/N's 4692001 thru 4692208)	3
Squat Switch (L/H side)	2
Up Limit Switch	2
Vacuum Regulator(s)	9

Access Plates and Panels
Figure 1 (Sheet 3 of 3)

CHAPTER

7

LIFTING AND SHORING

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CHAPTER 7

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
7-List of Effective Pages	1	Sep 15/09			
	2	Sep 15/09			
7-Table of Contents	1	Sep 15/09			
	2	Sep 15/09			
7-10-00	1	Sep 15/09			
	2	Sep 15/09			

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CHAPTER 7 - LIFTING AND SHORING

TABLE OF CONTENTS

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JACKING

1. General

Jacking the airplane is necessary to service the landing gear and perform other operations. The jacking operation can be performed by using tripod jacks; in other situations (emergency, post-accident lifting), slings or airbags should be used.

If wing or fuselage shoring is required, make sure the support is contoured to conform with the surface it is supporting.

2. Jacking

CAUTION: DISABLE LIFT DETECTOR, WINDSHIELD HEAT, AND PITOT HEAT PRIOR TO JACKING AIRCRAFT.

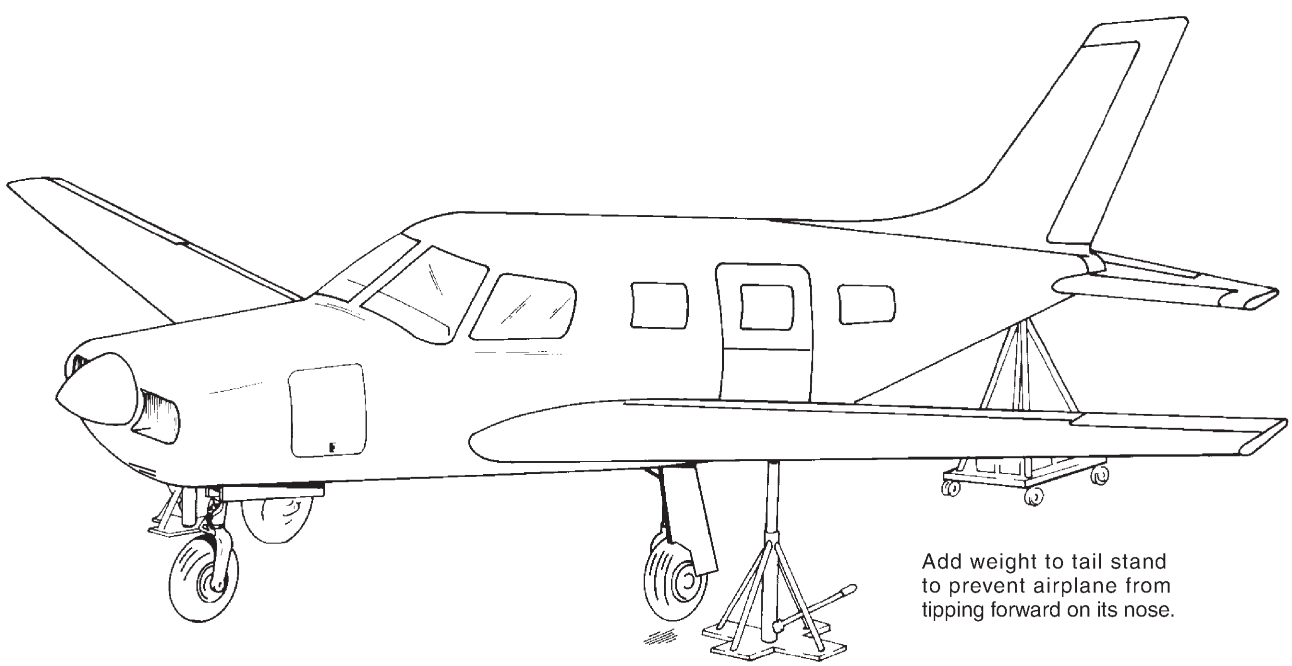
CAUTION: WHEN THE AIRCRAFT HAS OPTIONAL RADAR POD ON RIGHT WING ENSURE JACK DOES NOT COME IN CONTACT WITH RADAR POD

A. Align jacks under wing with respective pads on the wing main spar.

CAUTION: ENOUGH SUPPORT BALLAST MUST BE ADDED TO THE TAILSTAND TO PREVENT THE AIRPLANE FROM TIPPING FORWARD ON ITS NOSE SECTION. MAKE SURE TO ACCOUNT FOR SOMEONE IN THE FUSELAGE IF APPLICABLE.

B. Attach tail stand with approximately 400 pounds ballast to the tail skid.

C. Carefully raise jacks until all three wheels are clear of surface.



Jacking Arrangement
Figure 1

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CHAPTER

8

LEVELING AND WEIGHING

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CHAPTER 8

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
8-List of Effective Pages	1	Sep 15/09			
	2	Sep 15/09			
8-Table of Contents	1	Sep 15/09			
	2	Sep 15/09			
8-10-00	1	Sep 15/09			
	2	Sep 15/09			
8-20-00	1	Sep 15/09			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Sep 15/09			

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CHAPTER 8 - LEVELING AND WEIGHING

TABLE OF CONTENTS

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WEIGHING

1. Ballast

CAUTION: DO NOT REMOVE BALLAST WEIGHT AND TRAY ASSEMBLY, IF INSTALLED.

A ballast weight and tray assembly may have been installed at the factory under the ELT mounting tray. If so, this is permanent ballast and must not be removed.

NOTE: Any requirement for ballast adjustment based on field modifications should be determined per lastest revision of AC 43.13-1. The structure for additional ballast installation(s) should be structurally substantiated based on FAA-approved data.

2. Weighing

The airplane may be weighed by the following procedure:

A. Position a scale and a ramp in front of each of the three wheels.

B. Secure the scales from rolling forward, tow the airplane up onto the scales, and block the wheels.

C. Remove the ramp so as not to interfere with the scales.

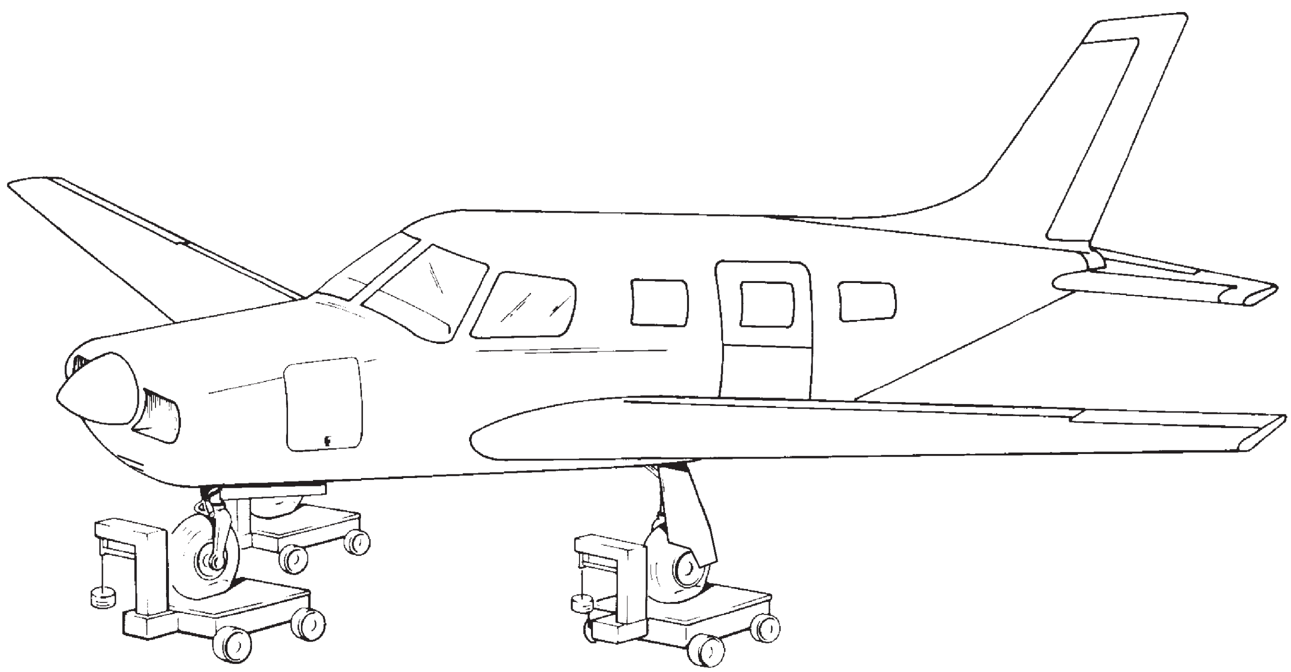
D. If the airplane is to be weighed for weight and balance computations, level the airplane per instructions given in the paragraph titled Leveling.

E. Fill engine with oil to full capacity.

F. Place pilot and copilot seats in fifth notch aft of forward position. Put flaps in the fully retracted position and all control surfaces in the neutral position. Tow bar should be in the proper location and all entrance and baggage doors should be closed.

G. Weigh the airplane inside a closed building to prevent errors in scale readings due to wind.

H. With the airplane level and brakes released, record the weight shown on each scale. Deduct the tare weight, if any, from each reading.



Weighing
Figure 2

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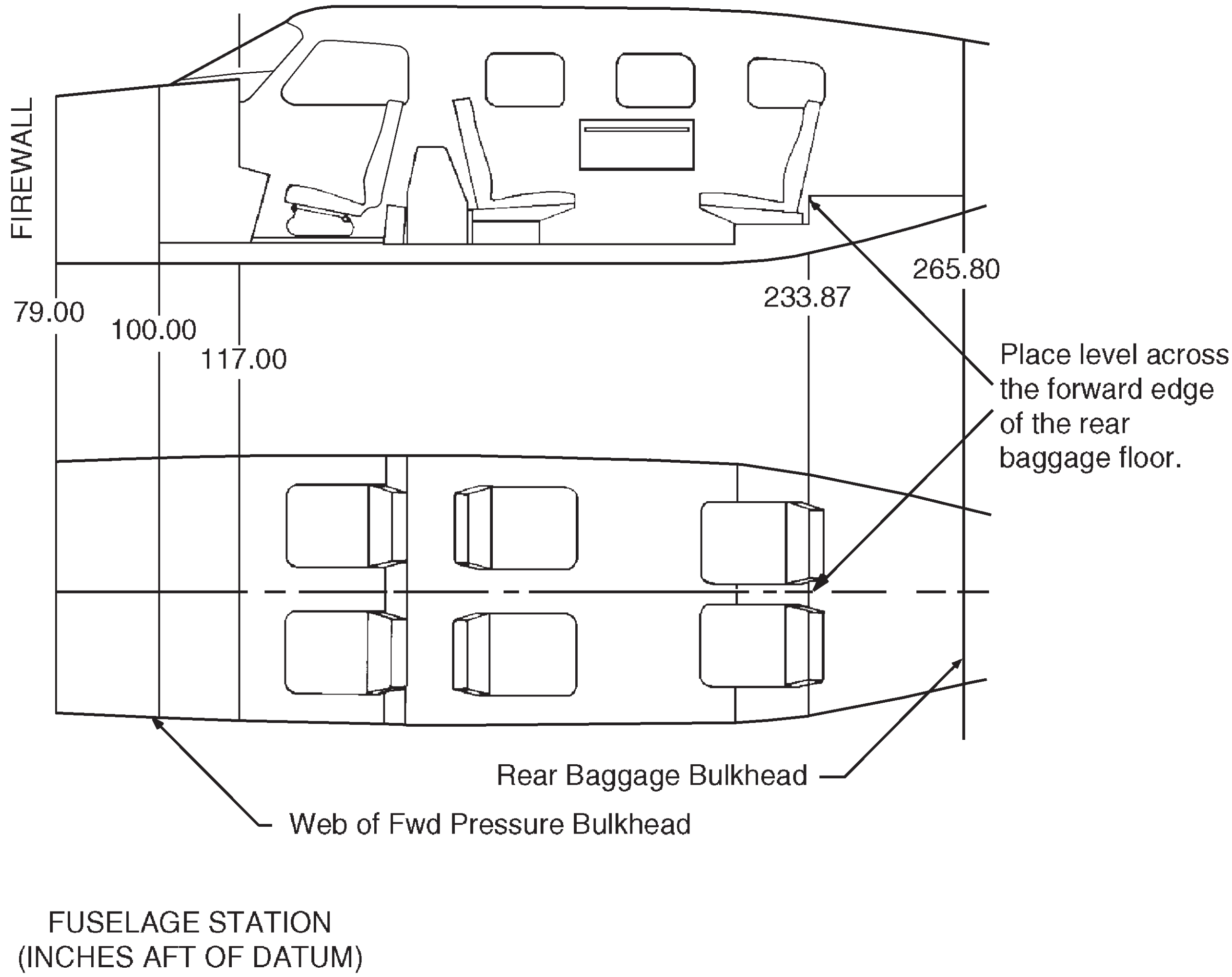
LEVELING

The airplanes are provided with a means for longitudinal and lateral leveling. The airplanes may be leveled while on jacks, during the weighing procedure while the wheels are on scales, or while the wheels are on the ground. To level the airplane for purposes of weighing or rigging, the following procedures may be used:

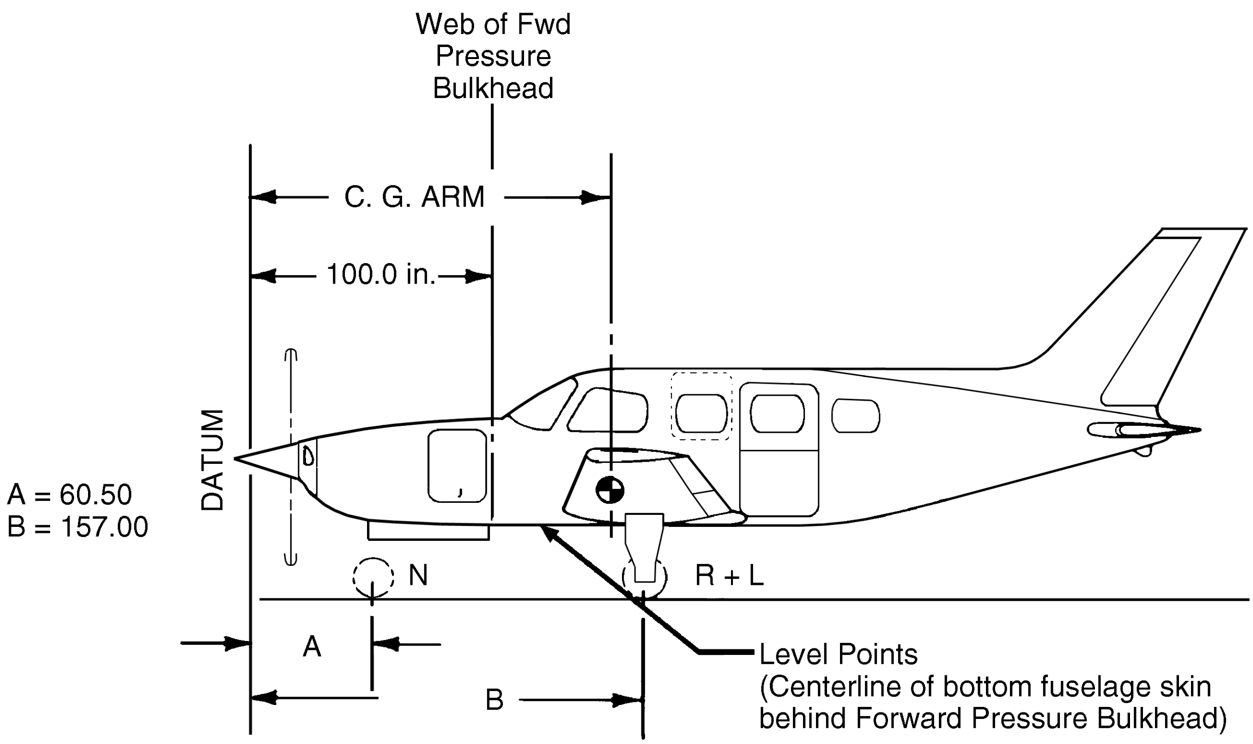
NOTE: Always level the airplane laterally first, then level the airplane longitudinally.

A. To laterally level the airplane, place a level across the forward edge of the rear baggage compartment. If the airplane is resting on its gear, raise or lower one wing tip by deflating the main gear tire on the high side of the airplane until the bubble of the level is centered. If the airplane is on jacks, adjust either wing jack until the bubble of the level is centered.



Leveling Laterally
Figure 1

B. To longitudinally level the airplane, place a level against the centerline of the bottom fuselage skin, behind the forward pressure bulkhead and adjust the jacks until the level is centered. Should the airplane be either on scales or on the floor, first insert a 3.4 inch spacer on each of the main gear struts and a 3.0 inch spacer on the nose gear strut. Level airplane by deflating (or inflating, as required) the nose wheel tire to center the bubble in the level.



Leveling Longitudinally
Figure 2

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CHAPTER

9

TOWING AND TAXIING

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CHAPTER 9

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
9-List of Effective Pages	1	Feb 21/20			
	2	Sep 15/09			
9-Table of Contents	1	Feb 21/20			
	2	Sep 15/09			
9-00-00	1	Sep 15/09			
	2	Sep 15/09			
9-10-00	1	Feb 21/20			
	2	Sep 15/09			
9-20-00	1	Sep 15/09			
	2	Sep 15/09			

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CHAPTER 9 - TOWING AND TAXIING

TABLE OF CONTENTS

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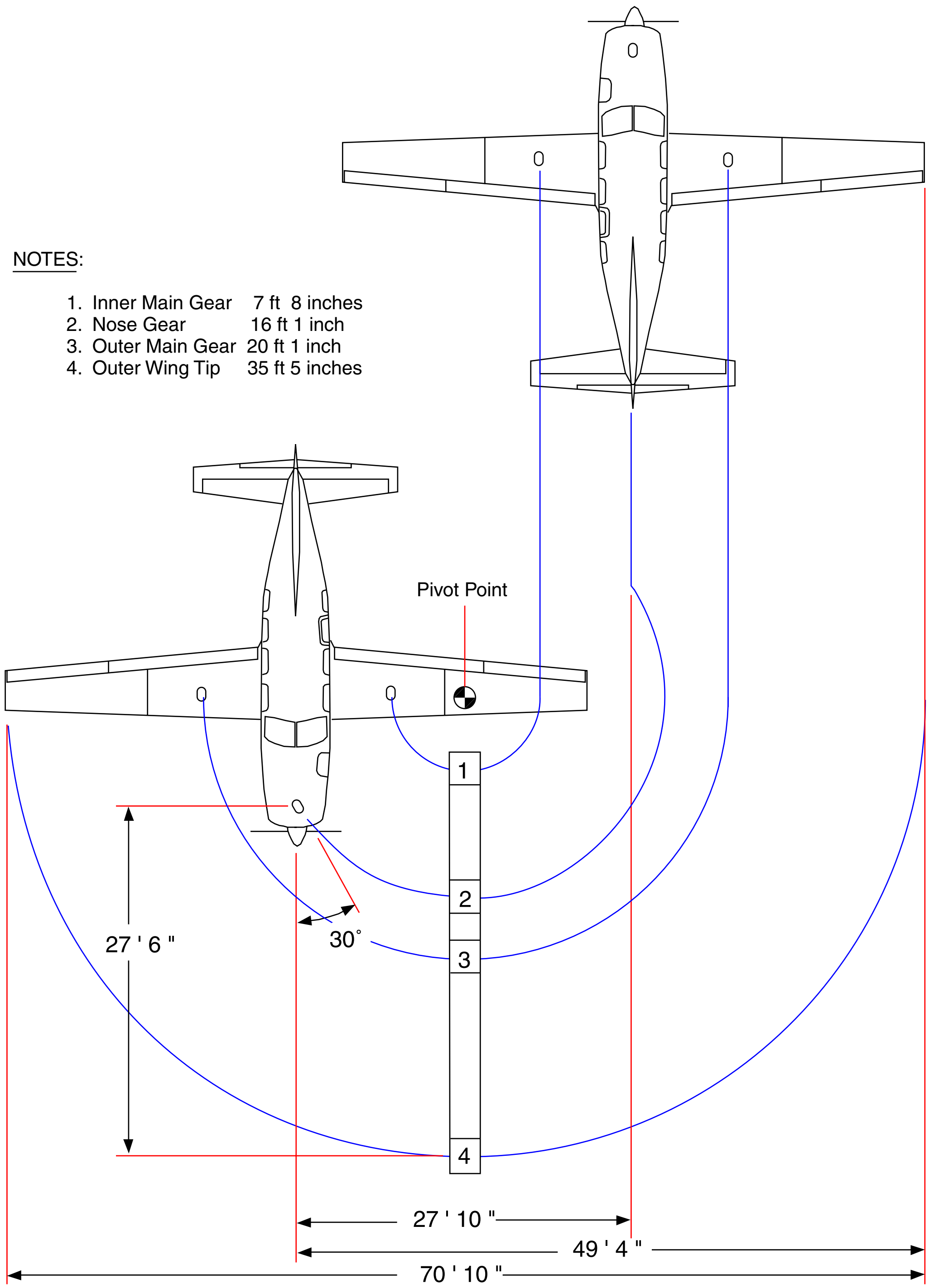
GENERAL

Before towing or taxiing airplane, ground personnel must be informed by a qualified pilot or other personnel about tow turning limits of nose gear, engine starting and shutdown procedures, and any other system functions required to properly and safely move the airplane.



Turning Radius and Limits
Figure 3

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TOWING

CAUTION: WHEN TOWING WITH POWER EQUIPMENT, USE ONLY EQUIPMENT THAT IS APPROPRIATE FOR THE AIRPLANE CLASS AND WEIGHT.

CAUTION: WHEN TOWING WITH POWER EQUIPMENT, DO NOT TURN THE NOSE GEAR IN EITHER DIRECTION BEYOND ITS STEERING RADIUS LIMITS AS THIS WILL RESULT IN DAMAGE TO THE NOSE GEAR AND STEERING MECHANISM.

CAUTION: IF AIRPLANE IS STUCK (OR THE WHEELS ARE OTHERWISE NOT FREEWHEELING), DO NOT TOW THE AIRPLANE FROM THE NOSE. ATTACH TOW LINES TO THE MAIN GEAR STRUTS AS DESCRIBED BELOW.

The airplane may be moved by using the nose wheel steering bar that is stowed in the forward baggage area or by using power equipment appropriate for the airplane class and weight (i.e., that will not damage or cause excess strain to the nose gear, steering assembly, engine mount, etc.). The stem on the bar is inserted in the hollow of the nose wheel axle at its right side.

In the event towing lines are necessary, lines (rope) should be attached to both main gear struts as high up on the tubes as possible. Lines should be long enough to clear the nose and/or tail by not less than 15 feet, and a qualified person should ride in the pilot’s seat to maintain control by use of the brakes.

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TAXIING

Before attempting to taxi the airplane, ground personnel should be checked out by a qualified pilot or other responsible person. Engine starting and shut-down procedures should be covered as well. When it is ascertained that the propeller back blast and taxi areas are clear, apply power to start the taxi roll and perform the following checks:

CAUTION: DO NOT OPERATE THE ENGINE AT HIGH RPM WHEN RUNNING UP OR TAXIING OVER GROUND CONTAINING LOOSE STONES, GRAVEL OR ANY LOOSE MATERIAL THAT MAY CAUSE DAMAGE TO THE PROPELLER BLADES.

CAUTION: OBSERVE WING CLEARANCES WHEN TAXIING NEAR BUILDINGS OR OTHER STATIONARY OBJECTS. IF POSSIBLE, STATION A GUIDE OUTSIDE THE AIRPLANE TO OBSERVE.

CAUTION: WHEN TAXIING ON UNEVEN GROUND, AVOID HOLES AND RUTS.

A. Taxi forward a few feet and apply brakes to determine their effectiveness.

B. Taxi with propeller set in low pitch, high RPM setting.

C. While taxiing, make slight turns to ascertain the effectiveness of steering.

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CHAPTER

10

PARKING AND MOORING

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CHAPTER 10

LIST OF EFFECTIVE PAGES

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CHAPTER 10 - PARKING AND MOORING

TABLE OF CONTENTS

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PARKING

1. Parking

When parking the airplane, ensure it is sufficiently protected against adverse weather conditions and presents no danger to other aircraft. When parking the airplane for any length of time or overnight, it must be moored according to sub-system section on mooring below.

NOTE: Care should be taken when setting brakes that are overheated, or when setting brakes during cold weather when moisture can freeze the brakes.

2. Locking Airplane

The cabin and baggage compartment doors are provided with a key lock on the outside. These locks use the same key, including the optional locking gas cap prior to 2006. For the optional locking gas cap original equipment and service replacements in 2006 and up, a separate key is provided.

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MOORING

CAUTION: WHEN MOORING, USE SQUARE OR BOWLINE KNOTS. DO NOT USE SLIP KNOTS.

The airplane is moored to ensure its security and protection under various weather conditions. To moor the airplane, use the following procedures:

A. Head airplane into wind, if possible.

B. Block wheels.

C. Lock aileron and elevator controls with front seat belts.

CAUTION: WHEN USING ROPE OF NON-SYNTHETIC MATERIAL, LEAVE SUFFICIENT SLACK TO AVOID DAMAGE TO AIRPLANE.

D. Secure wing tie-down ropes to retractable tie-down rings (located on underside of wings, outboard of the main gear) and to tail tie-down ring at approximately 45 degrees.

NOTE: Additional preparations for high winds include using tie-down ropes from the landing gear forks, and securing the rudder.

E. Install pitot tube cover.

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CHAPTER

11

PLACARDS AND MARKINGS

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CHAPTER 11

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
11-List of Effective Pages	1	Mar 25/21
	2	Sep 15/09
11-Table of Contents	1	Mar 25/21
	2	Sep 15/09
11-00-00	1	Feb 15/14
	2	Dec 15/14
	3	Dec 15/14
	4	Dec 15/14
11-20-00	1	Mar 25/21
	2	Mar 25/21
	3	Mar 25/21
	4	Mar 25/21
	5	Mar 25/21
	6	Nov 30/17
	7	Nov 30/17
	8	Nov 30/17
11-30-00	1	Nov 30/17
	2	Mar 25/21
	3	Dec 15/14
	4	Nov 30/17
	5	Dec 15/14
	6	Dec 15/14
	7	Dec 15/14
	8	Dec 15/14
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	16	Nov 30/17
	17	Mar 25/21
	18	Nov 30/17
	19	Nov 30/17
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	34	Nov 30/17
	35	Nov 30/17
	36	Nov 30/17
	37	Nov 30/17
	38	Nov 30/17

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CHAPTER 11 - PLACARDS AND MARKINGS

TABLE OF CONTENTS

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GENERAL

1. Decals

NOTE: See also, Placards, Nameplates, and Decals, below.

For proper adhesion of decals, all surfaces must be clean and free of wax, oil, etc. Porous surfaces must be sealed. Sand and clean rough surfaces to remove any residue.

A. Paper Decals

(1) Installation

Soak paper decals in water 1 to 3 minutes. Place one decal edge on receiving surface and slide decal off of paper backing. Blot water around decal with a soft absorbent cloth. Remove bubbles trapped beneath decal by wiping carefully towards nearest edge with a cloth. Coat decal with clear varnish to protect from deterioration and peeling.

(2) Removal

Remove paper decals by rubbing with cloth dampened with lacquer thinner. Use lacquer thinner sparingly if decals are applied over painted or doped surfaces.

B. Vinyl Film Decals

(1) Installation

Separate paper backing from vinyl film. Remove paper adhering to film by rubbing with a clean water saturated cloth or a piece of masking tape. Apply cyclohexanone or equivalent, to adhesive side of film. Position and apply decal while adhesive is still tacky. Work a roller across decal until all air bubbles are removed.

(2) Removal

To remove a vinyl decal, place cloth saturated with cyclohexanone or methyl ethyl ketone on decal. Scrape with micarta scraper. Remove remaining adhesive with cloth dampened with dry cleaning solvent.

C. Metal Decals

(1) Installation

(a) Cellophane backed.

1) Immerse in water 1 to 3 minutes.

2) Remove and dry.

3) Remove cellophane backing.

4) Position on receiving surface. (For large foil decals, position center or receiving surface and work outward from center.)

5) Roll with rubber roller and press all edges firmly.

(b) Paper backed.

1) Peel backing from decal.

2) Apply light coat of cyclohexanone.

3) Position and smooth per steps 4 and 5 of cellophane backed decals.

(c) Metal decals with no adhesive.

1) Apply cement MIL-A-5092 to decal and receiving surface.

2) Allow cement to dry until tacky.

3) Apply and smooth down decal.

4) Remove excess adhesive with aliphatic naphtha.

(2) Removal

To remove metal decals, moisten decal edge with aliphatic naphtha and peel off decal.

2. Placards, Nameplates, and Decals

(PIR-PPS65103, Rev. J.)

A. Self-Adhesive Installation

NOTE: For proper adhesion of decals, apply when the temperature is between 55° and 90° F.

(1) Wipe the area where the item is to be installed, using a clean cloth soaked with isopropyl alcohol. Dry with a clean, dry cloth.

NOTE: Dry the area before the alcohol evaporates, to avoid discoloration and deposits at the edges of the wetted area.

(2) Lift a corner or an edge of backing from the item, without touching the adhesive.

(3) Position as specified by the applicable figure in 11-20-00 or 11-30-00 and secure a free corner or edge.

(4) Using a squeegee or equivalent item to hold the item to the panel, draw the rest of the backing from the item.

(5) Press the item in place with the squeegee by wiping across it from end to end, around edges and corners, and into recessess using firm, deliberate strokes and heavy pressure.

(6) Remove the premask from face of item by using an eraser or finger nail to lift a corner. Then peel back as parallel to the item as possible.

NOTE: Any attempt to squeegee the item after the premask is removed will cause shiny marks, blemishing the item's background.

(7) If any portion of the item tends to lift off, remove and discard it. Repeat this procedure to install a replacement item.

B. Application of Edge Sealer (Decals only)

Decals intended for use in harsh environments have an overlay film applied during manufacturing. This overlay film typically extends 0.25 inch around the perimeter of the decal. If any of the overlay film is cut or otherwise damaged during installation, then an additional sealer must be applied along all edges of the decal. 3M 3950 Edge Sealer (P/N 279-474) is an acceptable sealer to be used in this application. Apply Edge Sealer as follows:

(1) Apply edge sealer to the edges of the decal by straddling the decal and the aircraft structure with a ¼-inch brush or dauber.

(2) Wipe this interface in a smooth continuous motion to provide a fillet-type seal that extends onto the decal and aircraft structure by aprroximately 0.1 inch.

(3) Ensure that the entire edge is sealed without gaps or air bubbles.

(4) Touch up as required with additional brush coats of edge sealer within 2 hours.

(5) Wipe all excess sealer with a dry wiper.

(6) Air dry or apply heat (up to 250˚F) to achieve a tack-free condition of the edge sealer.

C. Without Adhesive - Installation with Tape

(PIR-PPS-50085, Rev. B.)

To bond to the aircraft any placards, etc., that do not have an adhesive backing, use 3M™ VHB™ pressure sensitive acrylic foam tape, as follows.

(1) Confirm the correct location to adhere the double-sided tape. Place location marks (using masking tape) on the aircraft surface to precisely position the tape.

(2) Clean the surface using acetone or a 50:50 mixture of isopropyl alcohol and water. Wipe the area dry with a clean, lint-free, dry cloth.

(3) Wipe both faying surfaces with a brush or swab saturated with 3M Tape Primer 94 (P/N 279-127, one quart). Use the minimum amount of primer that will coat the desired surface areas. Allow 10 minutes for it to dry.

NOTE: Use a minimum coating of the tape primer. More is not better.

CAUTION: THE ADHESION OF THE TAPE IS ALMOST IMMEDIATE, SO USE EXTREME CARE AND APPLY VERY LITTLE CONTACT PRESSURE. AVOID THE NEED TO REPOSITION THE TAPE.

(4) Handle the double-sided acrylic foam tape by its edges only; carefully position the tape using the location marks. Adhesion is almost immediate, so use extreme care and as little contact pressure as possible. Avoid the need to reposition the tape, but if necessary, move the tape immediately and cautiously.

(5) When the placement of the tape is completed, use a roller, squeegee, or other smoothing tool to apply even pressure against the tape. Pay close attention to the tape edges.

CAUTION: DO NOT TRY TO REPOSITION THE TAPE AFTER USING SMOOTHING TOOL. DO NOT PULL ON THE TAPE OR ATTEMPT TO LIFT OR PEEL ITS EDGES.

(6) Keep the outward liner intact on the tape and position the item to be bonded with the tape over the location. Remove the tape liner using the pull tabs, folding them back at an approximately 45-degree angle from under the item, exposing the tape to the item.

(7) Once bonded, if there is some bowing or deformation of the adhered item, apply pressure not exceeding 5 psi with a shot bag (approximately 6 inches wide, 36 lbs per foot is only 1 psi) for 30 minutes.

3. Meyercord / Ink Transfers (Decals)

(PIR-PPS65104, Rev. J.)

Transfers (decals) installed on the instrument panel are Meyercord / Ink type. The following procedures should be followed in the event one or more of these transfers (decals) must be replaced.

A. Removal

CAUTION: DO NOT USE LACQUER THINNER ON ANY PANEL THAT HAS BEEN PAINTED WITH ENAMEL OR LACQUER. INSTRUMENT PANELS ARE PAINTED AT THE FACTORY WITH POLYURETHANE PAINTS.

Remove placard to be replaced with a clean cloth dampened with lacquer thinner.

CAUTION: BUTYL GLYCOL ETHER/ISOPROPYL ALCOHOL SOLVENT SOLUTION WILL REMOVE ENAMEL, LACQUER, AND POLYUTHERANE BASED PAINT PRODUCTS IF LIQUID IS DROPPED ONTO PAINTED SURFACE AND NOT REMOVED IMMEDIATELY.

If panel is painted with enamel or lacquer use a clean cloth dampened with Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution to remove placard to be replaced.

B. Installation

(1) If required, clean surface to receive transfer (decal) using alcohol. Newly painted surfaces need no preparation.

(2) Mix a solution consisting of six (6) parts water and one (1) part Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution (P/N 179-497).

(3) Submerge transfer (decal) in the mixed Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution for approximately 3 to 5 seconds.

(4) Remove transfer (decal) from mixed solvent solution and lay in position.

(5) Using a plastic squeegee, squeegee out from center to edges to remove excess solution.

(6) Wait approximately 30 to 60 seconds, then slide the backing paper off and wipe up the excess solution with a damp cloth.

(7) Wait at least 30 minutes at room temperature before wiping the face of the transfer (decal) with a damp cloth to remove excess solvent residue.

(8) Allow the transfer (decal) to dry thoroughly, (tack free in 2 hours at room temp) before handling.

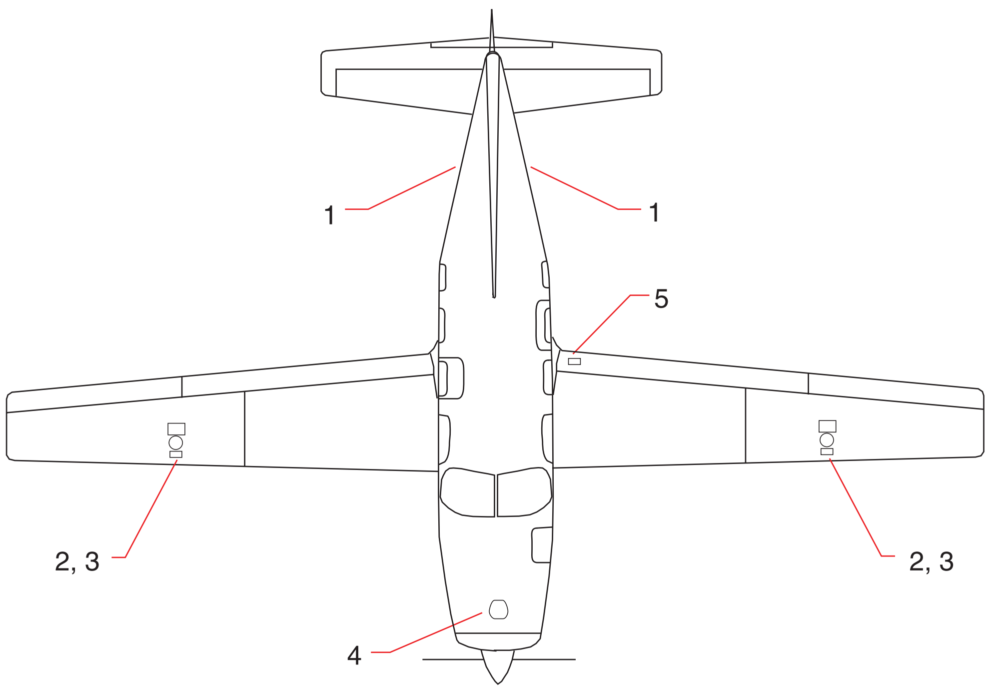
EXTERIOR PLACARDS AND MARKINGS

The airplane nameplate placard (i.e. - Item 7, Figure 1) is located below the tailcone at approximately F.S. 311.50. The placard identifies the airplane by its model number and serial number. Should a question arise concerning the care of the airplane, it is important to include the airplane serial number in any correspondence to Piper Aircraft, Inc.

NOTE: Any time an airplane is repainted or touched up, inspect all placards to ensure that they are not covered with paint, are legible, and securely attached.

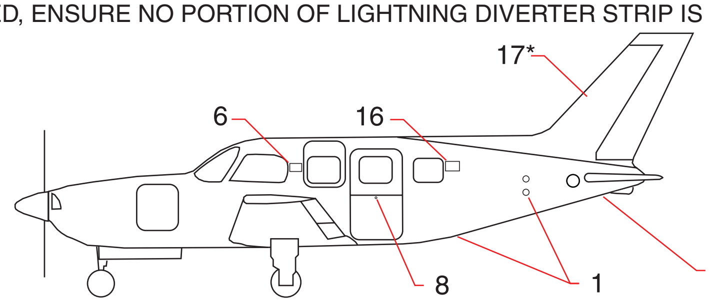
NOTE: Item numbers refer to Decal and Placard detail views shown in Sheets 3 thru 6.

89502 BB
105825 BB



TOP VIEW

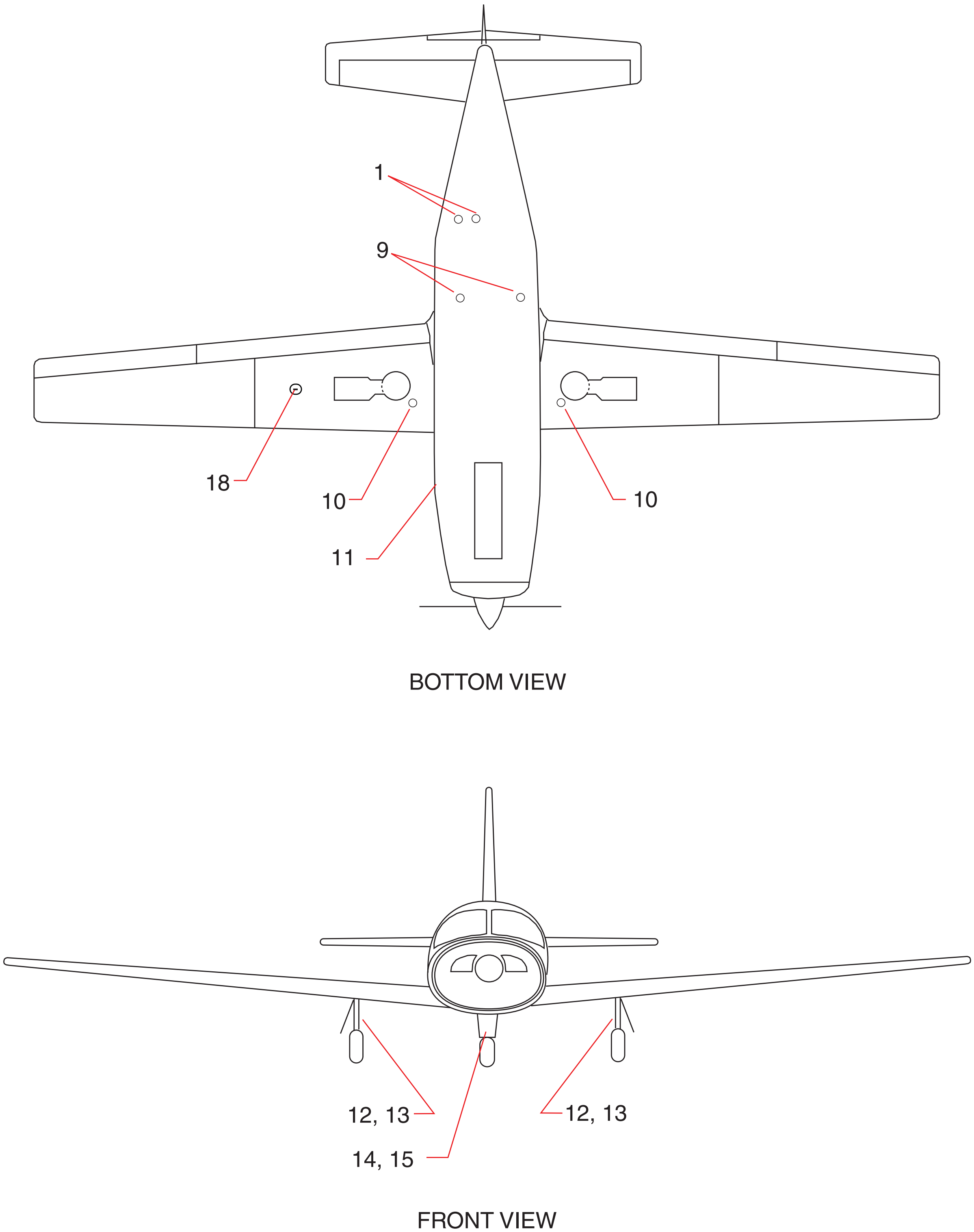
* IF EQUIPPED, ENSURE NO PORTION OF LIGHTNING DIVERTER STRIP IS COVERED.



SIDE VIEW

Placards and Markings - Exterior
Figure 4 (Sheet 1 of 8)

NOTE: Item numbers refer to Decal and Placard detail views shown in Sheets 3 thru 6.

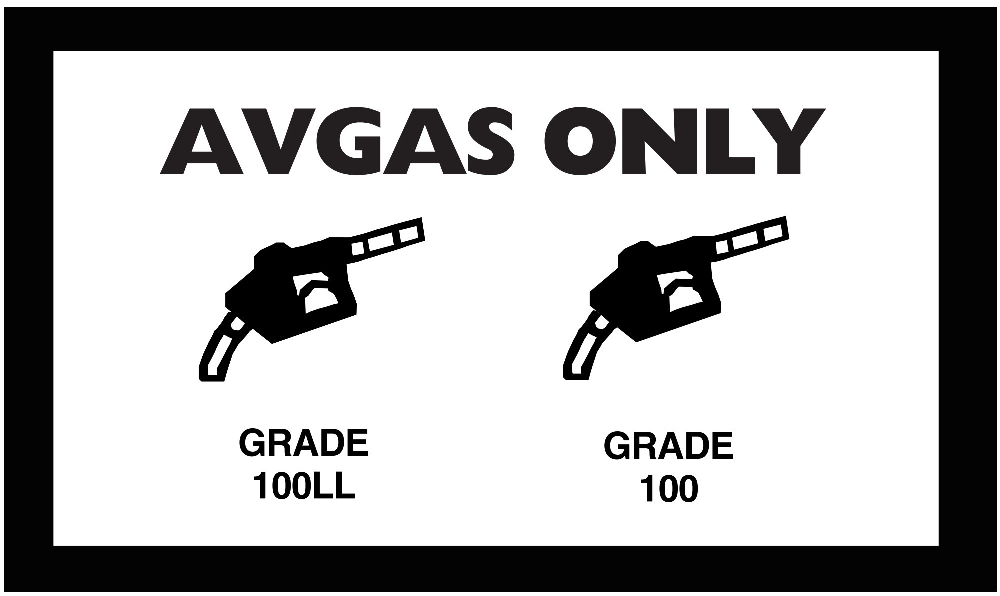


Placards and Markings - Exterior
Figure 1 (Sheet 2 of 8)

STATIC VENTS
KEEP CLEAN
100700-004 or 100700-005

(REF)

Decal #1
(100700-004 or 100700-005)



Decal #2
(456-682)

(REF)

DURING FUELING AND DEFUELING OPERATIONS,
ATTACH GROUNDING CABLE TO GROUND
ATTACH PIN LOCATED ON MAIN STRUT.

Decal #3
(100700-006 or 100700-007)

Placards and Markings - Exterior
Figure 1 (Sheet 3 of 8)

* RECOMMENDED SAE GRADE OIL

AVERAGE AIR TEMPERATURE	MIL-L-22851 OR SAE J1899 ASHLESS DISPERSANT
ALL TEMPS	15W-50W OR 20W-50W
ABOVE 80° F	60
ABOVE 60° F	40 OR 50
30° F TO 90° F	40
0° F TO 70° F	30, 40 OR 20W-40
BELOW 10° F	30 OR 20W-30

* REFER TO LATEST REV OF LYCOMING S.I. 1014
OIL CAPACITY 12 QTS (11.3 L)

Placard #4
(89379-002)

NO
STEP

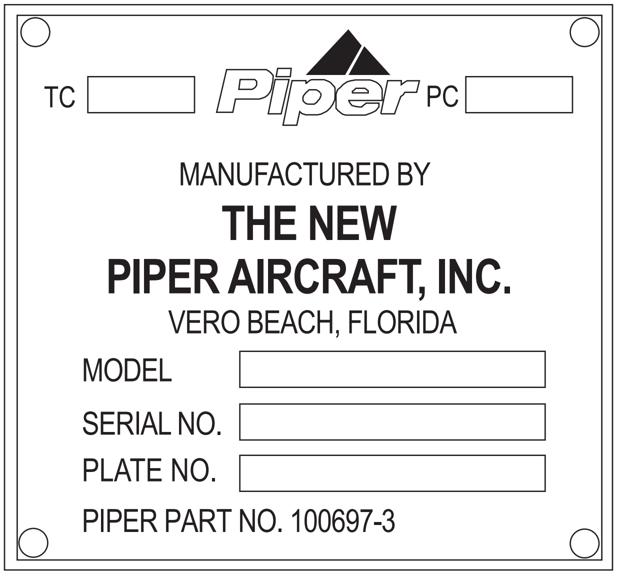
Placard #5
(100700-002 or 100700-003)

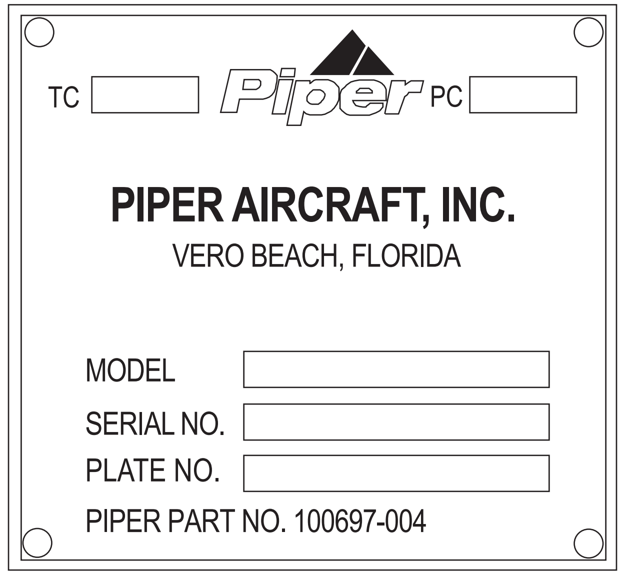
Stencil #6A
(89910-002)
(S/N's 4636001 thru 4636363)

Stencil #6B
(105030-002)
(S/N's 4636364 & UP)

Stencil #6C
(105852-002)
(S/N's 4692001 & UP)

Placards and Markings - Exterior
Figure 1 (Sheet 4 of 8)

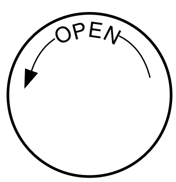




Placard #7

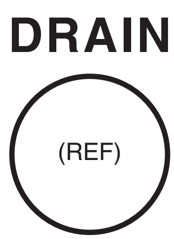
(S/N's 4636001 thru 4636374)

(S/N's 4636375 & UP)
(S/N's 4692001 & UP)

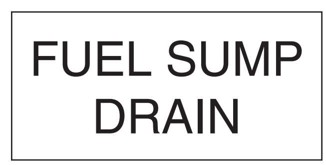


Decal #8
(100700-014)

NOTE: Not required if door handle has etched arrow and lettering.



Decal #9
(100700-017)



Decal #10
(100700-008 or 100700-009)

Decal #11
(100700-010 or 100700-011)

Decal #12
(100700-016)

Placards and Markings - Exterior
Figure 1 (Sheet 5 of 8)

Placard #13
(87369-080) Main Gear

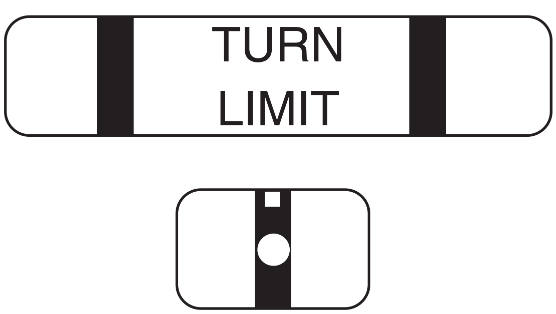
STRUT AND TIRE
SERVICE INSTRUCTIONS

PLACE AIRPLANE ON JACKS AND EXTEND STRUT COMPLETELY. RELEASE AIR AND REMOVE AIR VALVE. SLOWLY COMPRESS THE STRUT. SLOWLY FILL COMPRESSED STRUT WITH MIL-H-5606 HYDRAULIC FLUID THROUGH THE FILLER OPENING UNTIL IT REACHES THE TOP OF THE FILLER PLUG HOLE. MANUALLY COMPRESS AND EXTEND STRUT SEVERAL TIMES TO REMOVE TRAPPED AIR. ADD FLUID EACH TIME IF REQUIRED. REPLACE VALVE AND, WITH MAIN GEAR STILL CLEAR OF GROUND, INFLATE STRUT 250 PSI AIR PRESSURE. INFLATE MAIN TIRE TO 55 PSI AIR PRESSURE. REMOVE JACKS FROM AIRPLANE (NORMAL MAIN GEAR STRUT EXTENSION IS 3.40 INCHES WITH APPROX. 1560 LBS. STATIC LOAD ON EACH MAIN GEAR).

Placard #14
(87369-075) Nose Gear

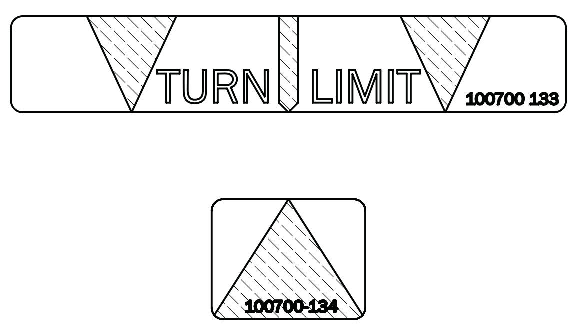
STRUT AND TIRE
SERVICE INSTRUCTIONS

PLACE AIRPLANE ON JACKS AND EXTEND STRUT COMPLETELY. RELEASE AIR AND REMOVE AIR VALVE. SLOWLY COMPRESS THE STRUT. SLOWLY FILL COMPRESSED STRUT WITH MIL-H-5606 HYDRAULIC FLUID THROUGH THE FILLER OPENING UNTIL IT REACHES THE TOP OF THE FILLER PLUG HOLE. MANUALLY COMPRESS AND EXTEND STRUT SEVERAL TIMES TO REMOVE TRAPPED AIR. ADD FLUID EACH TIME IF REQUIRED. REPLACE VALVE AND, WITH NOSE GEAR STILL CLEAR OF GROUND, INFLATE STRUT TO A PRESSURE OF 110 PSI. INFLATE NOSE GEAR TIRE TO 50 PSI AIR PRESSURE. REMOVE JACKS FROM AIRPLANE (NORMAL NOSE GEAR STRUT EXTENSION IS 1.70 INCHES WITH APPROX. 720 LBS. STATIC LOAD ON NOSE GEAR).



Original equipment in
S/N's 4692001–4692214
S/N's 4636001–4636688
Top - (35669-008)
Bottom - (35669-005)

Decal #15



S/N's 4692215 and Up; S/N's 4636689 and Up
(Authorized service replacement in S/N's 4692001–4692214 and S/N's 4636001–4636688, see SB 1293 latest revision)
Top - (100700-133); Bottom - (100700-134)

Placards and Markings - Exterior
Figure 1 (Sheet 6 of 8)

Decal #16
656-400
(S/N's 4636349 thru 4636363 only)

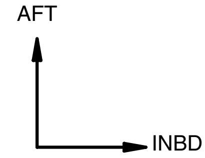
Stencil or Decal #17
Piper Logo (106760-002, -003, -007)
(Typical)

Placards and Markings - Exterior
Figure 1 (Sheet 7 of 8)

USE ONLY NON-MAGNETIC FASTENERS
WITHIN 12 INCHES OF MAGNETOMETER
100700-103

LOOKING UP

Original Equipment in
S/N's 4636633, 4636652–4636694



USE ONLY NON-MAGNETIC TOOLS AND
APPROVED FASTENERS
WITHIN 12 INCHES OF MAGNETOMETER
100700-111

LOOKING UP

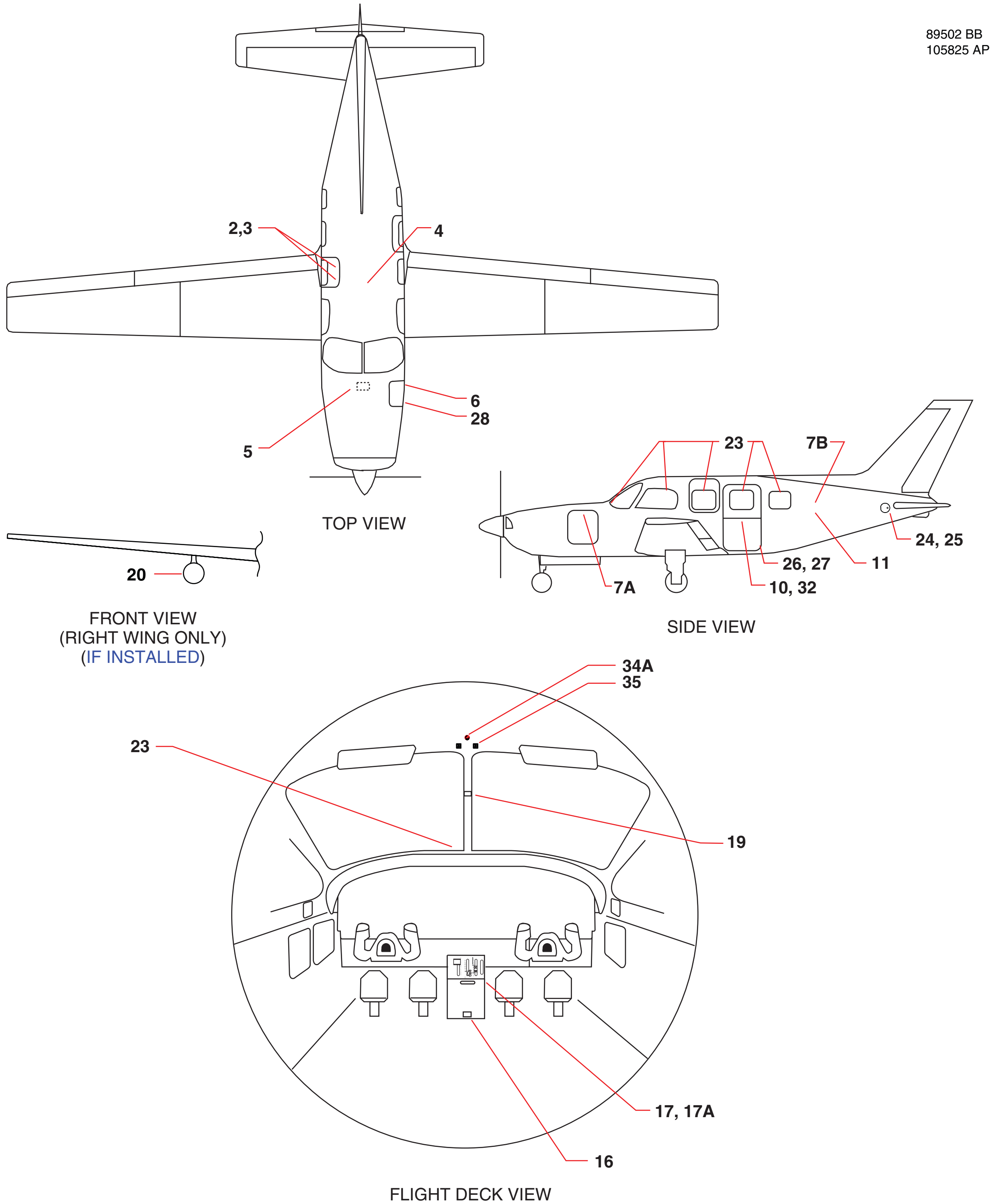
S/N's 4636695 and Up
(Authorized service replacement in
S/N's 4636633, 4636652–4636694)

DECAL #18

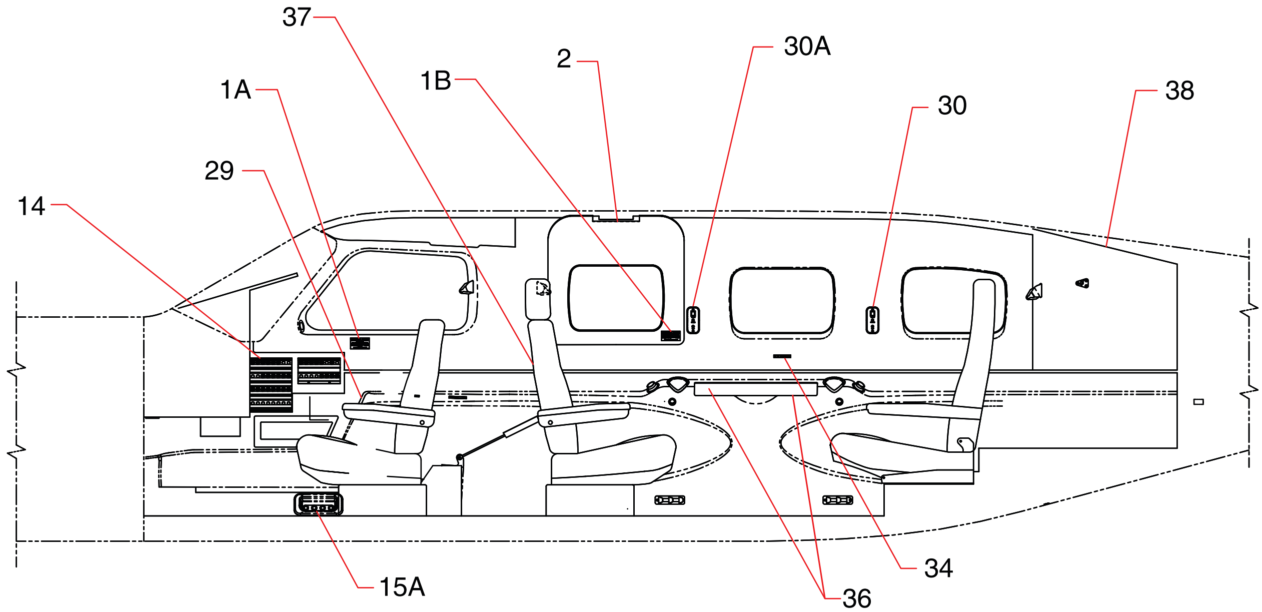
Placards and Markings - Exterior
Figure 1 (Sheet 8 of 8)

INTERIOR PLACARDS AND MARKINGS

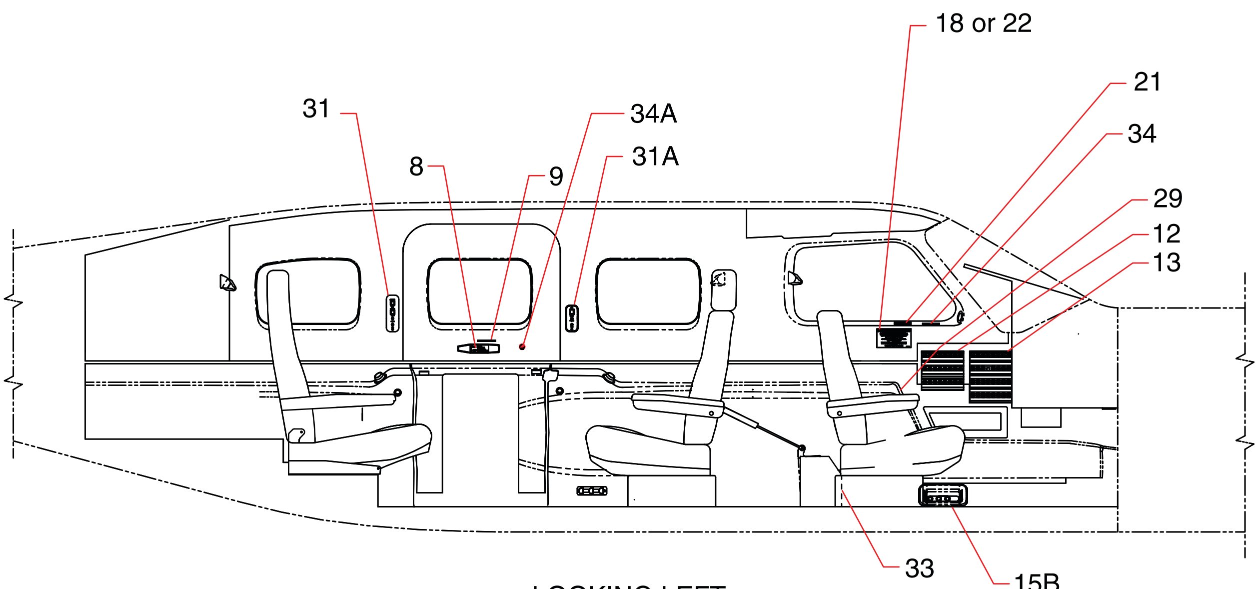
NOTE: Item numbers refer to Decal, Transfer, and Placard detail views shown in later Figures.



Placards and Markings - Interior - Overview
Figure 1 (Sheet 1 of 2)



LOOKING RIGHT



LOOKING LEFT

Placards and Markings - Interior - Overview
Figure 1 (Sheet 2 of 2)

11302

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Decal #1A (Cockpit)
(100700-062) (Shown)
(S/N's 4636001 and up)

EMERGENCY OXYGEN
IN DRAWER UNDER SEAT
(AISLE ACCESS)
PULL MASK OUT OF DRAWER FULLY
AT FULL EXTENSION GIVE CORD A TUG
MAXIMUM DURATION ------- 15 MINS

SEE POH
NO SMOKING WHILE IN USE
100700-062

Decal #1B (Cabin)
(83998-002 - S/N's 4636001 to 4636374)
(100700-048 - S/N's 4636375 and up)

EMERGENCY OXYGEN
IN DRAWER UNDER SEAT

PULL MASK OUT OF DRAWER FULLY
AT FULL EXTENSION GIVE CORD A TUG
MAXIMUM DURATION ------- 15 MINS

SEE POH
NO SMOKING WHILE IN USE

EMERGENCY EXIT

REMOVE GLASS
PULL DOOR IN - LIFT UP

Decal #2

(PA-46-350P
84987-004 Assy.
or
PA-46R-350T
106729-002 - Top
and
106730-002 - Bottom)

TABLE MUST BE STOWED
DURING TAKE-OFF & LANDING

Decal #3
(100700-018)

FIRE
EXTINGUISHER

Decal #4
(84967-042)

FLUID LEVEL LINE

BRAKE FLUID RESERVOIR

FILL TO FLUID LEVEL LINE WITH PETROLEUM
BASE (RED) HYDRAULIC OIL *

* MIL-H-5606 OR EQUIVALENT

Decal #5
Top - (96548-000) Bottom - (96546-000)

Decals not required with new brake reservoir circa 2013.
See appropriate Parts Catalog to replace damaged or lost decals on existing brake reservoirs only.

Placards and Markings - Interior
Figure 2

EXTERNAL POWER
28 VOLTS D.C.
TURN MASTER SWITCH
AND ALL EQUIP. OFF
BEFORE INSERTING
OR REMOVING PLUG.

EXTERNAL POWER
28 VOLTS D.C.
TURN MASTER SWITCH
AND ALL EQUIP. OFF
BEFORE INSERTING
PLUG.

Decal #6
(100700-012)
(S/N's 4636001 to 4636374)

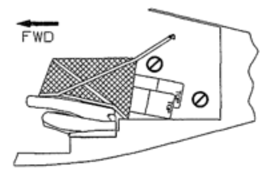
(100700-071)
(S/N's 4636375 and up)
(S/N's 4692001 and up)

MAXIMUM BAGGAGE THIS COMPARTMENT 100 LBS.

Decal #7A
Nose - (100700-013)
(S/N's 4636001 and up)
(S/N's 4692001 and up)

STANDARD CARGO NET INSTALLED
MAX BAGGAGE THIS COMPARTMENT
100 LBS (45.3 KG)

GOLF CARGO NET INSTALLED



⊘ NO BAGGAGE PERMISSIBLE
IN THESE AREAS WHEN THE
GOLF NET IS IN USE

MAX GOLF CLUB LOAD
105 LBS (47.6 KG)

MAX ADDITIONAL BAGGAGE
BEHIND LEFT SEAT
50 LBS (22.6 KG)

Decal #7B
Rear - (100700-038)
(S/N's 4636446 and up)
(S/N's 4692001 and up)

OPEN — PULL LATCH
ROTATE HANDLE UP

CLOSE — ROTATE HANDLE DOWN
ENSURE PIN WINDOWS GREEN

Decal #8
(84967-020)
(Mounted on Handle)

OPEN ⟷ CLOSE   ENSURE PIN
WINDOWS GREEN

Decal #9
(100700-015)

CAUTION
DO NOT ATTEMPT
TO CLOSE DOOR
WITH HANDLE IN
LATCHED POSITION

Decal #10
(100700-027)

REAR PASSENGER/BAGGAGE AREAS

MAXIMUM ALLOWABLE WEIGHT

MAXIMUM ALLOWABLE COMBINED WEIGHT IN AFT SEATS IS
POUNDS

LOAD IN ACCORDANCE WITH
WEIGHT AND BALANCE DATA

Decal #11
(101116-002)
(S/N'S 4636001 thru 4636557,
S/N'S 4692001 thru 4692183)

Placards and Markings - Interior
Figure 3

POSITION	STROBE	LANDING	TAXI	ICE

EXTERIOR LIGHTS

LTNG PROT	INSTR PANEL	ANNUN	AVIONICS DIMMING

INTERIOR LIGHTS

VENT DEFOG	PROP HEAT	STALL HEAT	PITOT HEAT	SURFACE DE-ICE	WNDSHLD HEAT CONTROL	POWER

ICE PROTECTION

(PA-46-350P - Typical)
(89844)

VENT DEFOG	STALL HEAT	L PITOT HEAT	R PITOT HEAT	PROP HEAT	SURFACE DE-ICE	WNDSHLD HEAT CONTROL	POWER

ICE PROTECTION

LTNG PROT	CABIN PRESSURE DUMP	VACUUM	LEFT FUEL LEVEL	RIGHT FUEL LEVEL

GEA	PFD 1 FAN	PFD 1	AUDIO MKR	ADC 1	AHRS 1	COM 1	INTEG AV 1

Decal **#12**
Pilot Side - Aft
(S/N'S 4636460, 4636463 and up, w/ G1000 - Typical,
S/N'S 4692134 and up, w/ G1000 - Typical)
(105555)

Placards and Markings - Interior
Figure 4

EMERG	L BOOST	R BOOST	FUEL QTY	FUEL FLOW	ELEC TAC	ENGINE GAGES	ENGINE MONITOR
FUEL PUMPS							
ENGINE SYSTEMS							

TURN COORD	LANDING GEAR INDICATION	LANDING GEAR WARNING	HYDRAULIC PUMP CONTROL	HYDRAULIC PUMP POWER	STALL WARN	FUEL MGT
FLIGHT SYSTEMS						

ALTERNATOR NO. 1	ALTERNATOR NO. 2	MAP LIGHT	FLAP MOTOR	FLAP WARN	HOUR METER

CABIN PRESS DUMP	ENGINE START	CO-PILOT TURN COORD	CABIN LIGHTS	CABIN FANS	AIR CONDITIONING CONTROL	AIR CONDITIONING POWER

(PA-46-350P - Typical)
(89844)

ENGINE START	ALTERNATOR NO. 1	ALTERNATOR NO. 2	LANDING	TAXI	NAV	STROBE	ICE
			EXTERIOR LIGHTS				

EMERG	L BOOST	R BOOST	ANNUN	AVIONICS DIMMING	INSTR PANEL	CABIN	MAP
FUEL PUMPS			INTERIOR LIGHTS				

HYDRAULIC PUMP CONTROL	HYDRAULIC PUMP POWER	LANDING GEAR INDICATION	LANDING GEAR WARN	STALL WARN	SPEED BRAKES	FLAP MOTOR	FLAP WARN
FLIGHT SYSTEMS							

PITCH PRESS	AUTOPILOT	STBY ATTITUDE INDICATOR	INVERTER	AIR CONDITIONING CONTROL	AIR CONDITIONING POWER	CABIN FANS	HOUR METER

Decal **#13**
Pilot Side - Forward
(S/N'S 4636460, 4636463 and up, w/ G1000 - Typical,
S/N'S 4692134 and up, w/ G1000 - Typical)
(105555)

Placards and Markings - Interior
Figure 5

DME 1	SPEAKER AMP	NAV 1	COMM 1	COMM 2	NAV 2	HEAD SET AMP	DME 2
RMI	ENCODER 1	XPDR 1	ADF 1	ADF 2	XPDR 2	ENCODER 2	GPS
PITCH TRIM	AUTO PILOT	COMPASS	RADAR ALT			STORM SCOPE	RADAR
		AVIONICS COOLING	LORAN	BUS TIE	FLITE PHONE	INV	

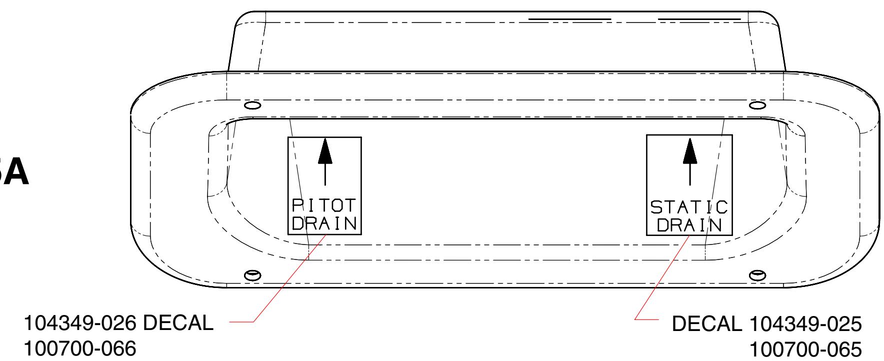
(PA-46-350P - Typical)
(89844)

RADAR	XPNDR 1	MFD	AV/MFD FAN	KEYPAD	GMC	XM
AVIONICS BUS TIE	INTEG AV 2	COM 2	AHRS 2	ADC 2		LTNG PROT
PFD 2	PFD 2 FAN	STORM SCOPE	TAS	DME	ADF	XPNDR 2

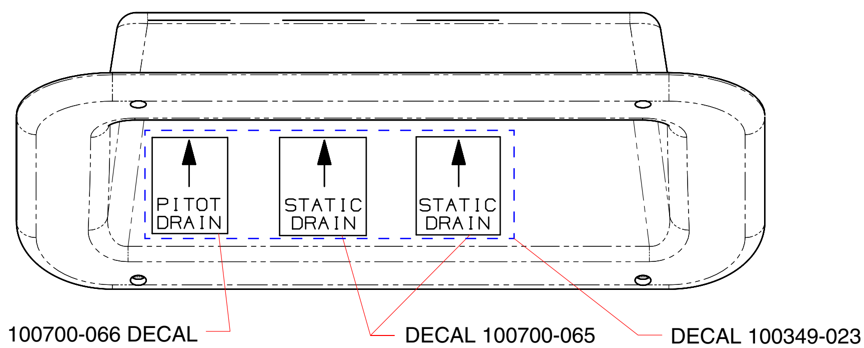
Decal **#14**
Copilot Side
(S/N'S 4636460, 4636463 and up, w/ G1000 - Typical,
S/N'S 4692134 and up, w/ G1000 - Typical)
(105555)

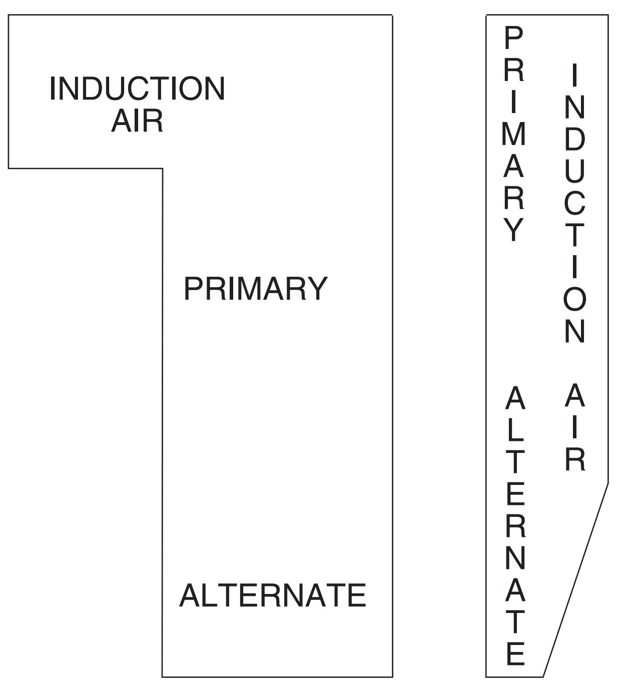
Placards and Markings - Interior
Figure 6

15A



15B



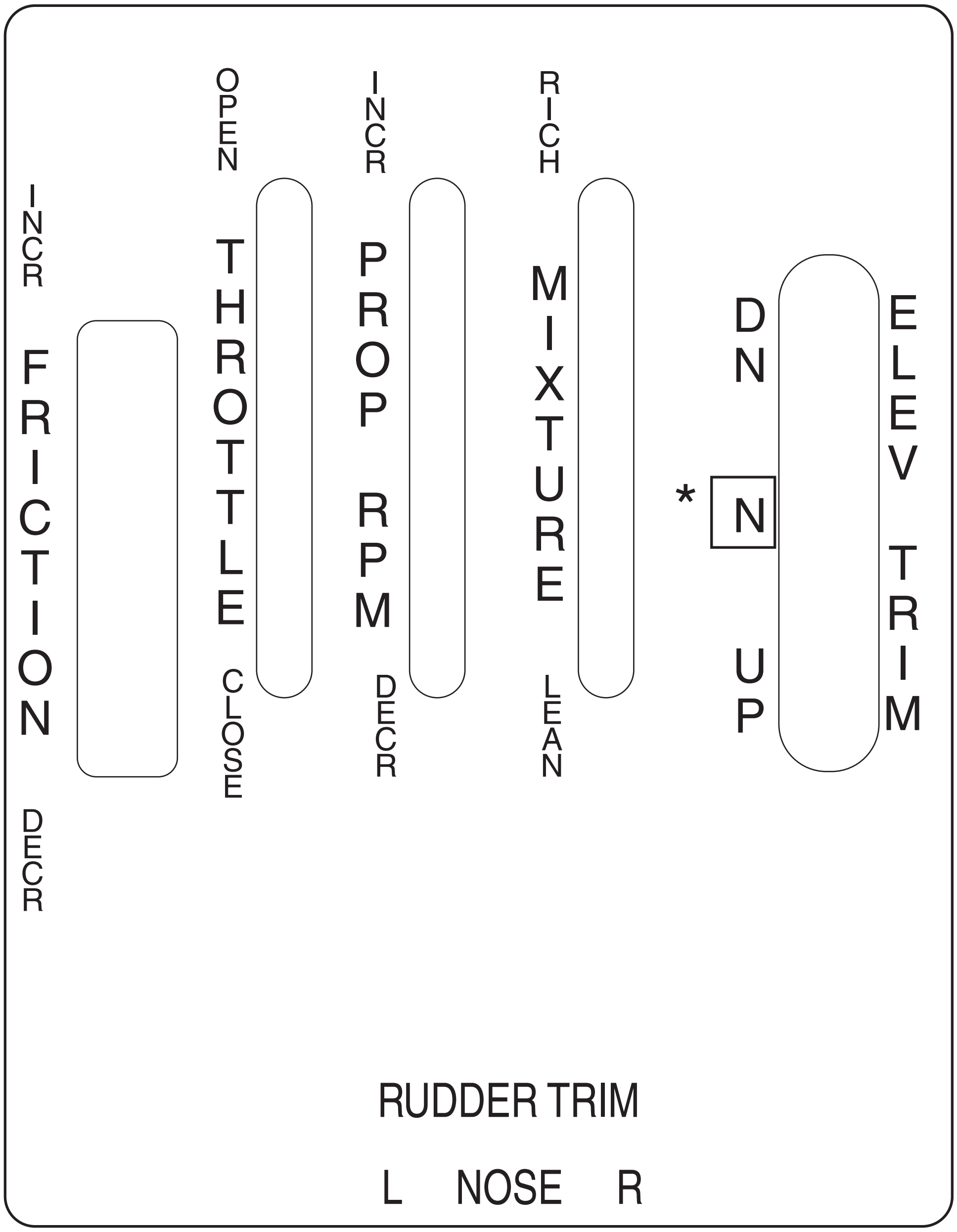


(84967-010)

(84967-348)
w/ G1000

Decal #16

Placards and Markings - Interior
Figure 7



Placard #17
(89969-002)

* Placard #17A
(100700-077)

Placards and Markings - Interior
Figure 8

THE MARKINGS AND PLACARDS INSTALLED IN THIS AIRPLANE CONTAIN OPERATING LIMITATIONS WHICH MUST BE COMPLIED WITH WHEN OPERATING THIS AIRPLANE IN THE NORMAL CATEGORY. OTHER OPERATING LIMITATIONS WHICH MUST BE COMPLIED WITH WHEN OPERATING THIS AIRPLANE IN THIS CATEGORY ARE CONTAINED IN THE AIRPLANE FLIGHT MANUAL. NO AEROBATIC MANEUVERS, INCLUDING SPINS, APPROVED.
THIS AIRCRAFT APPROVED FOR V.F.R., I.F.R., DAY AND NIGHT ICING FLIGHT WHEN EQUIPPED IN ACCORDANCE WITH THE AIRPLANE FLIGHT MANUAL.

WARNING

AIR CONDITIONER MUST BE OFF TO INSURE NORMAL TAKEOFF CLIMB PERFORMANCE.

WARNING

TURN OFF STROBE LIGHTS WHEN IN CLOSE PROXIMITY TO GROUND OR DURING FLIGHT THROUGH CLOUD, FOG OR HAZE.

PRESSURIZED LANDING NOT APPROVED.

Placard #18
(89579-002)

CAUTION

COMPASS
CAL. MAY
BE IN ERROR
WITH ELECT.
EQUIPMENT
OTHER THAN
AVIONICS ON.
87369-27

(87369-27)

STANDBY COMPASS

FOR CORRECT READING CHECK:
WINDSHIELD HT SWITCH OFF
PROP DE-ICE SWITCH OFF
COOLING SYSTEM OFF
COCKPIT AND CABIN HEATING OFF

(100700-041)
(with Avidyne)

STANDBY COMPASS

FOR CORRECT READING CHECK:
AVIONICS ON
PITOT HEAT OFF
AIR COND OFF
WINDSHIELD HEAT OFF
PROP HEAT OFF
STALL WARN HEAT OFF
AUXILIARY CABIN HT OFF

(100700-083)
(with Garmin G1000)

Decal #19

MAX WEIGHT 5 LBS (2.2 KG)

Decal #20
(100700-037)
(Mounted inside radome door.)

ALTERNATE STATIC SOURCE
(LOCATED PILOTS SIDE BELOW PANEL)
UP - ALTERNATE
DOWN - PRIMARY
(SEE POH SECTION 5 FOR ALTERNATE
STATIC SYSTEM CORRECTION)

(100700-064)
(S/N's 4636375 and up)

ALTERNATE STATIC SOURCE
(LOCATED PILOTS SIDE BELOW PANEL)
UP - ALTERNATE
DOWN - PRIMARY

(100700-063)
(S/N's 4692001 and up)

Decal #21 (Typical)
(Mounted below pilot's side window.)

See Figure 18, Decal #P4, for S/N's 4636001 thru 4636374

Placards and Markings - Interior
Figure 9

THIS AIRCRAFT MUST BE OPERATED AS A NORMAL CATEGORY AIRPLANE
IN COMPLIANCE WITH THE OPERATING LIMITATIONS STATED IN THE
FORM OF PLACARDS, MARKINGS AND MANUALS.
NO ACROBATIC MANEUVERS, INCLUDING SPINS, APPROVED.
THIS AIRCRAFT APPROVED FOR V.F.R., I.F.R., DAY AND
NIGHT ICING FLIGHT WHEN EQUIPPED IN ACCORDANCE
WITH THE AIRPLANE FLIGHT MANUAL.
WARNING
AIR CONDITIONER MUST BE OFF TO INSURE NORMAL
TAKEOFF CLIMB PERFORMANCE.
WARNING
TURN OFF STROBE LIGHTS WHEN IN CLOSE PROXIMITY
TO GROUND OR DURING FLIGHT THROUGH CLOUD, FOG
OR HAZE.
PRESSURIZED LANDING NOT APPROVED

Decal #22 - Mirage/M350 (Typical)
(100700-073)
(S/N's 4636375 and up)

THIS AIRCRAFT MUST BE OPERATED AS A NORMAL CATEGORY AIRPLANE
IN COMPLIANCE WITH THE OPERATING LIMITATIONS STATED IN THE
FORM OF PLACARDS, MARKINGS AND MANUALS.
NO ACROBATIC MANEUVERS, INCLUDING SPINS, APPROVED.
THIS AIRCRAFT APPROVED FOR V.F.R., I.F.R., DAY AND
NIGHT ICING FLIGHT WHEN EQUIPPED IN ACCORDANCE
WITH THE AIRPLANE FLIGHT MANUAL.
WARNING
AIR CONDITIONER MUST BE OFF TO INSURE NORMAL
TAKEOFF CLIMB PERFORMANCE.
WARNING
TURN OFF STROBE LIGHTS WHEN IN CLOSE PROXIMITY
TO GROUND OR DURING FLIGHT THROUGH CLOUD, FOG
OR HAZE.

Decal #22 - Matrix (Typical)
(100700-075)
(S/N's 4692001 and up)

FOR INSTALLATION ON UNPRESSURIZED AIRCRAFT ONLY

Decal #23
(100700-069)
(S/N's 4692001 and up)
Do not install on left hand Pilot window if
equipped with Heated windshield

Placards and Markings - Interior
Figure 10

OXYGEN - DANGER

NO SMOKING WHILE CHARGING

AVOID CONTACT WITH
OIL, SOAP, GREASY,
AND FATTY MATERIALS

USE ONLY AVIATOR
BREATHING OXYGEN

FILL TO 1850 PSI
MAX WORKING PRESSURE

DO NOT EXCEED 2200 PSI

Decal #24
Top - (100700-074) Bottom - (100700-070)
(S/N's 4692001 and up)

MAXIMUM WEIGHT
THIS COMPARTMENT
4 LBS (1.8 KG)
100700-039

MAXIMUM WEIGHT THIS
COMPARTMENT 4 LBS (1.8 KG)
100700-039

Placard #25
(100700-039)
(S/N's 4636375 and up)

COURTESY
LIGHT RESET

(84967-105)
(S/N'S 4636460, 4636463 and up, w/ G1000,
S/N'S 4692134 and up, w/ G1000)

BAGGAGE

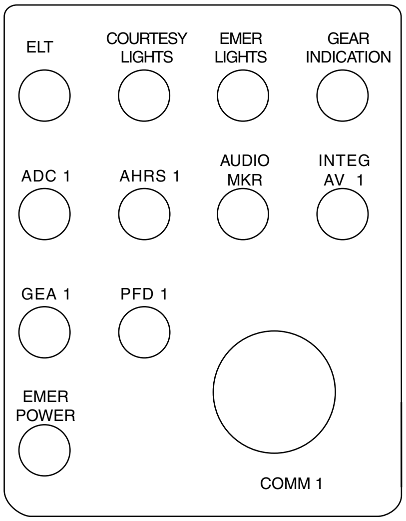
Decal #26
(84967-264)
(S/N'S 4636375 and up, w/ Entegra,
S/N'S 4692001 and up, w/ Entegra)

COURTESY
LIGHT RESET
100700-087

Decal **#27**
(100700-087)
(S/N'S 4636514 and up, w/ G1000,
S/N'S 4692173 and up, w/ G1000
and Service replacement for 84967-105)

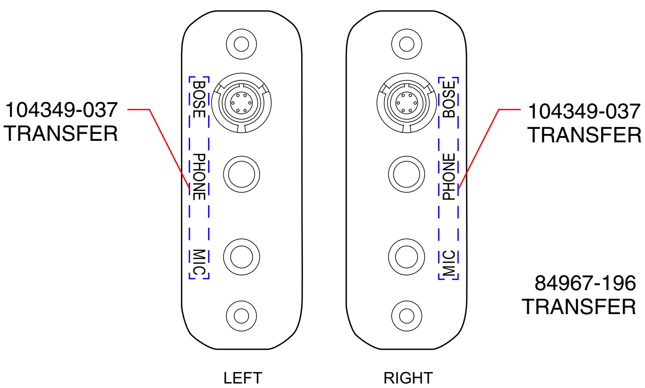
(Also service replacement for 84967-264,
S/N'S 4636446 and up, w/ Entegra,
S/N'S 4692001 and up, w/ Entegra)

Placards and Markings - Interior
Figure 11

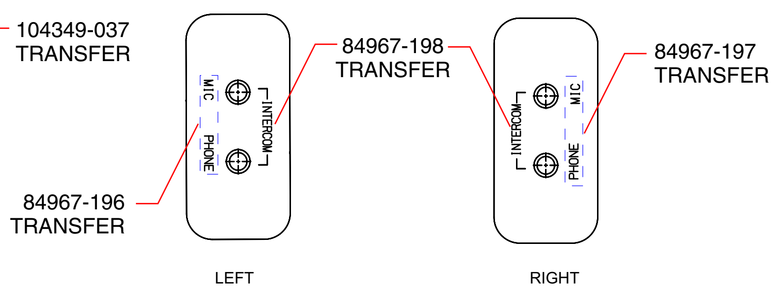


Placard #28 (Typical)
(100881-011 shown)

106277 H



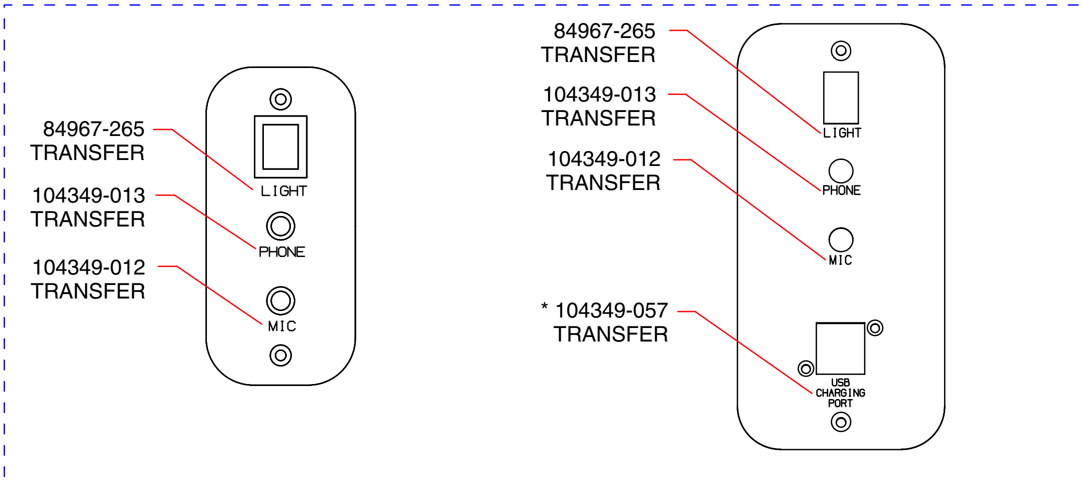
#29
WITH BOSE HEADSET OPTION



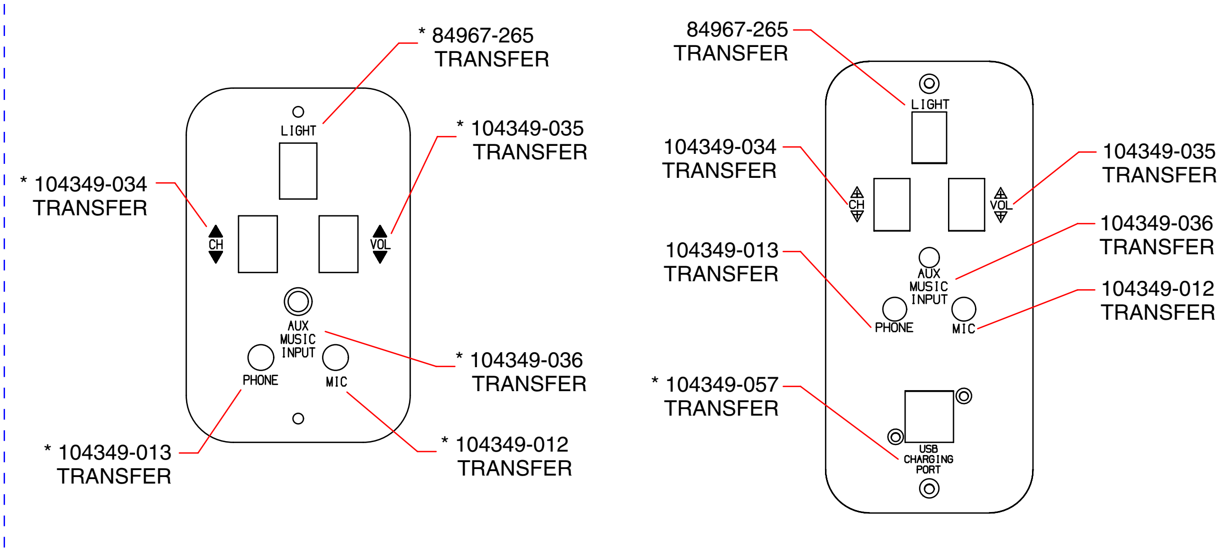
#29

Placards and Markings - Interior
Figure 12

113014



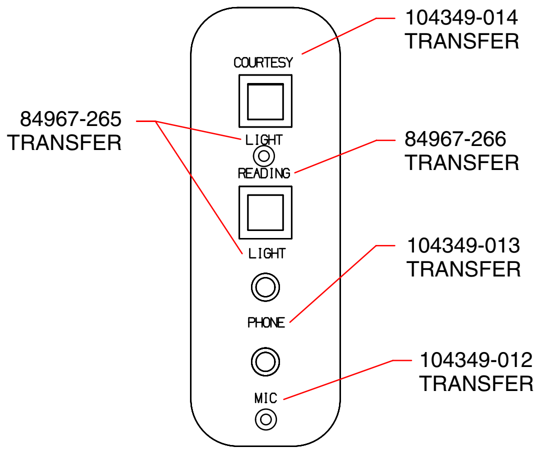
#30A
(1 of 2)



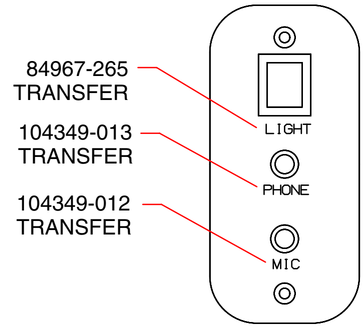
#30
(1 of 4)

(* OPTIONAL)

Placards and Markings - Interior
Figure 13



#31



#31A

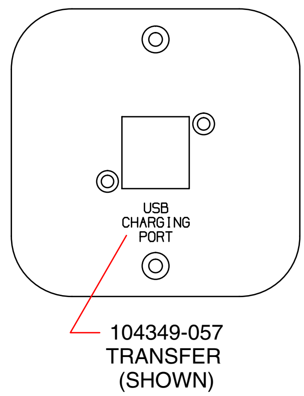
PILOTS, PASSENGERS AND BAGGAGE AREAS
MAXIMUM ALLOWABLE COMBINED WEIGHT
POUNDS

LOAD IN ACCORDANCE WITH
APPROVED WEIGHT AND BALANCE DATA

101116-004 PLACARD

#32

S/N's 4636558 and up, S/N's 4692184 and up

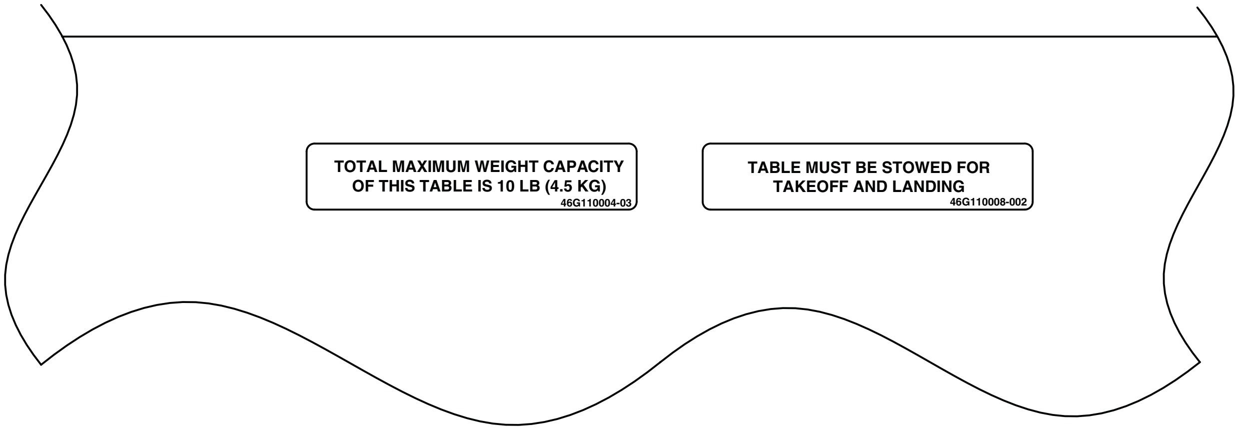


#33

S/N's 4636633, 4636652 and up

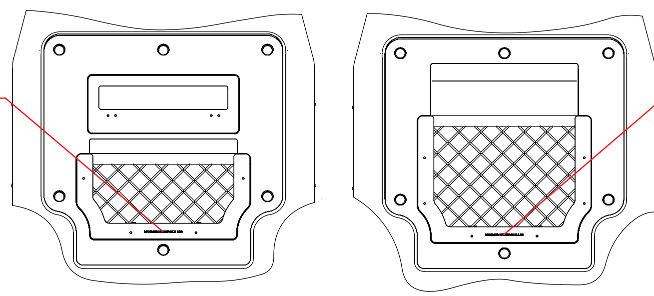
Placards and Markings - Interior
Figure 14 (Sheet 1 of 2)

113016

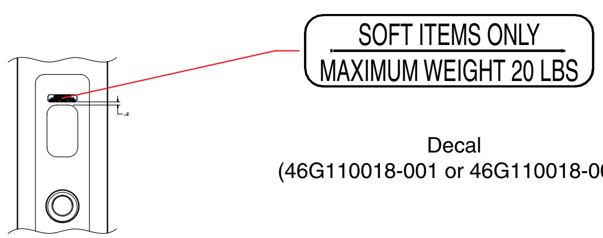


#36
EXECUTIVE TABLE

VIEW LOOKING DOWN
TABLE SHOWN EXTENDED



#37
AFT FACING SEATS



#38
CLOTHING ROD (If Installed)

Placards and Markings - Interior
Figure 14 (Sheet 2 of 2)

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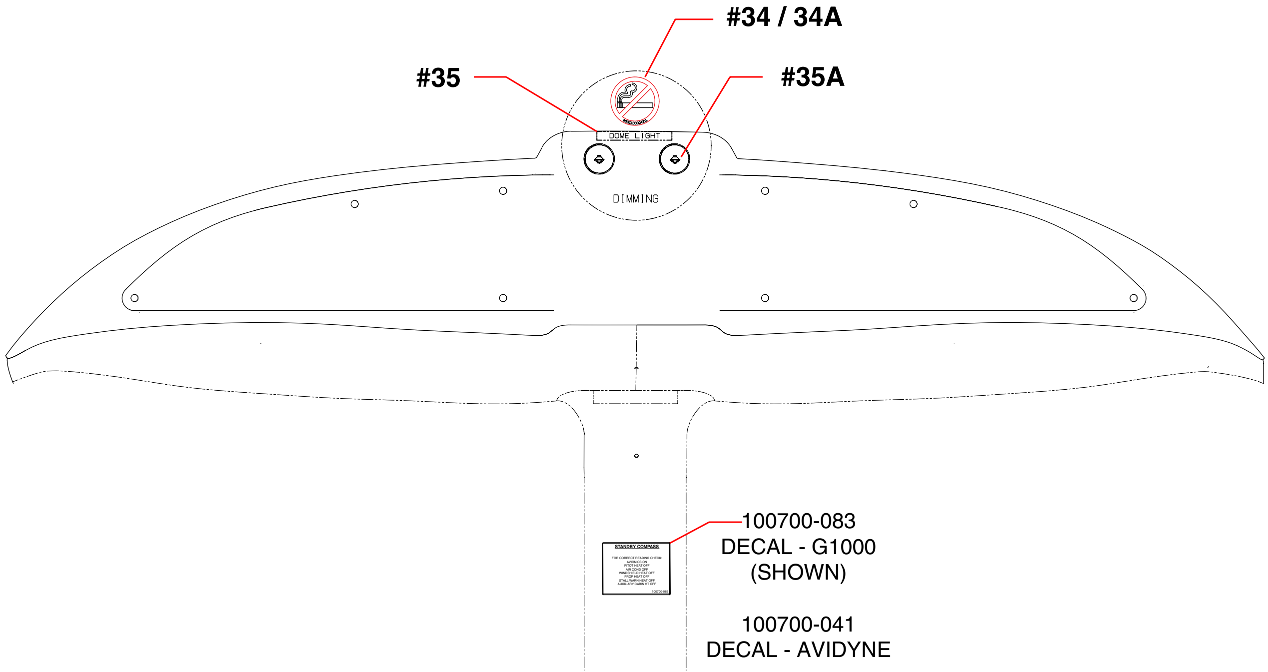
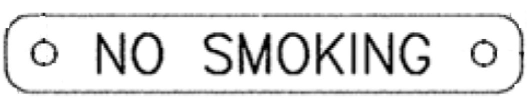
105825 AW 4



LOOKING FORWARD



100882-002 PLACARD

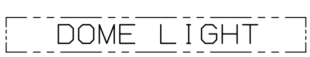
#34

(S/N's 4636001–4636685;
4692001 and up)

46G110002-001 DECAL

#34A

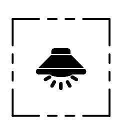
(S/N's 4636686 and up)



104349-001
TRANSFER
(WHITE LETTERS)
104349-002
TRANSFER
(BLACK LETTERS)

#35

(S/N's 4636001–4636715,
4692001–4692214)

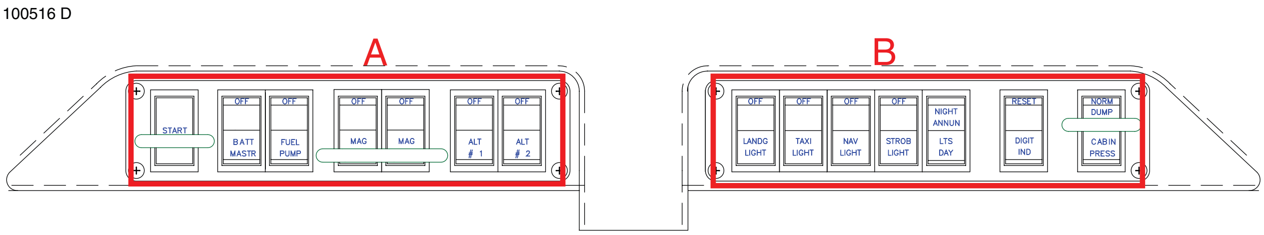


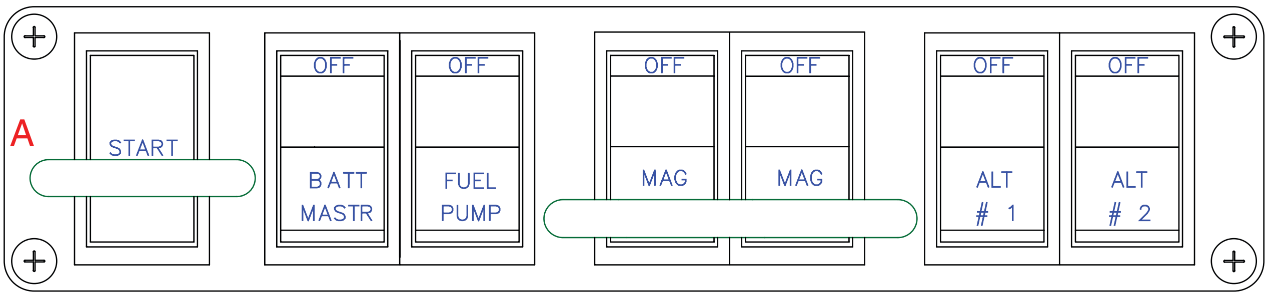
104349-065 (READING LIGHT)
TRANSFER

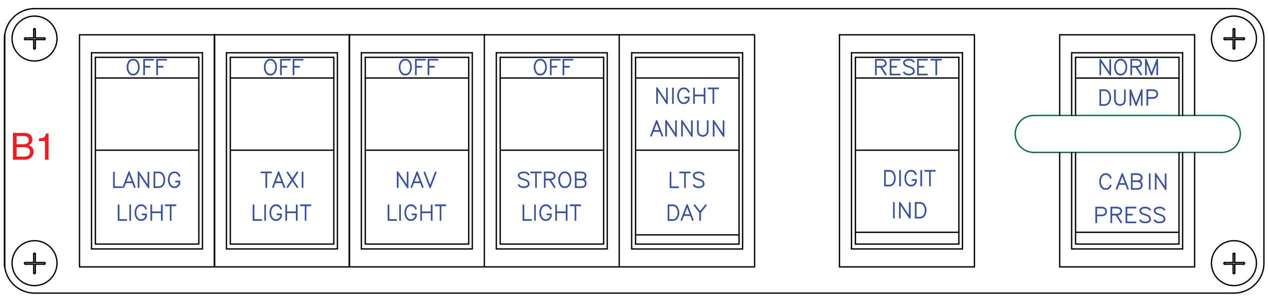
#35A

(S/N's 4636716 and up)

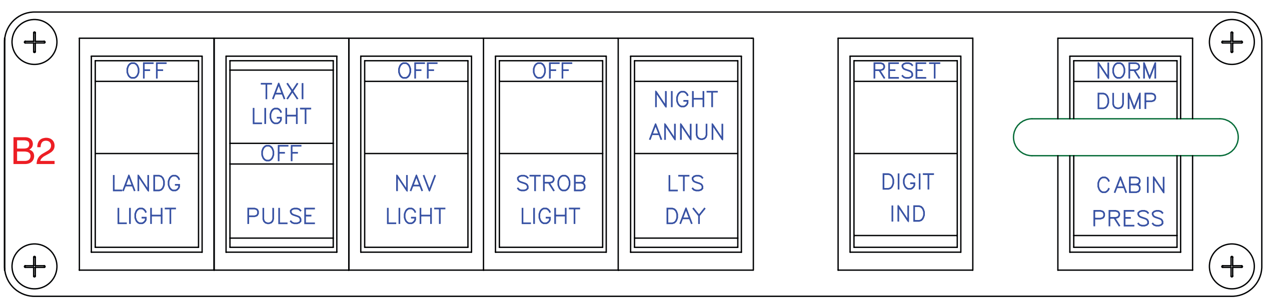
Placards and Markings - Overhead Switch Panel
Figure 15



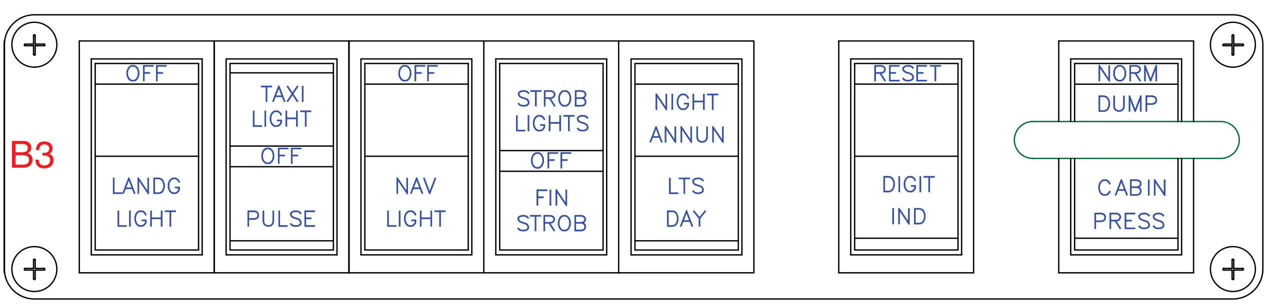




(S/N's 4636001 thru 4636186)



(S/N's 4636187 thru 4636313, less 4636299)

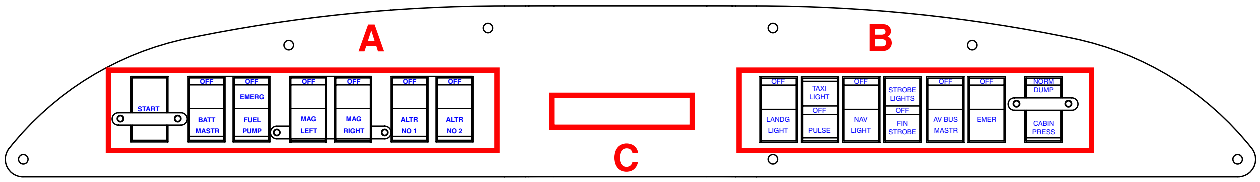


(S/N's 4636299, 4636314 and up - with conventional gauges or single-side EFIS)

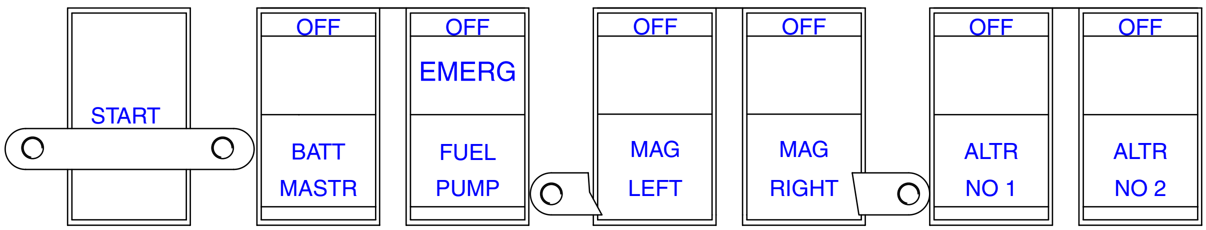
Effectivity
4636001 thru 4636374

Placards and Markings - Overhead Switch Panel
Figure 16

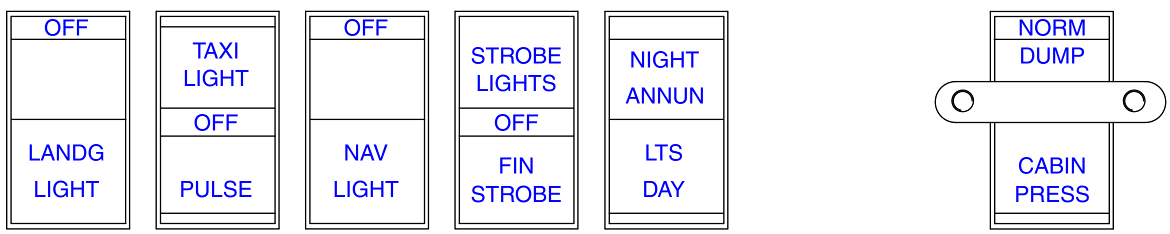
105116 V
105818 G



A

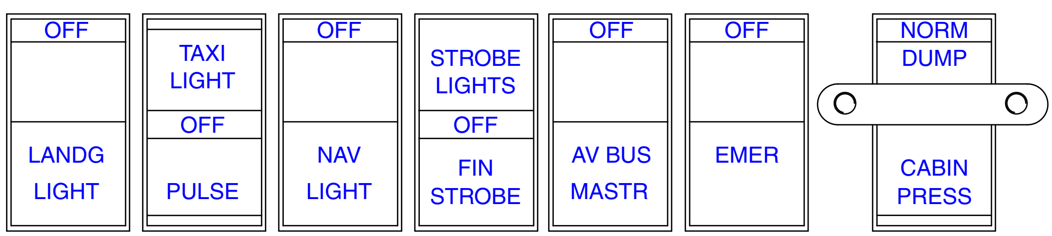


B4



(TYPICAL - S/N's 4636375 and up - with Avidyne)

B5



(TYPICAL - S/N's 4636460, 4636463 and up - with G1000)

C

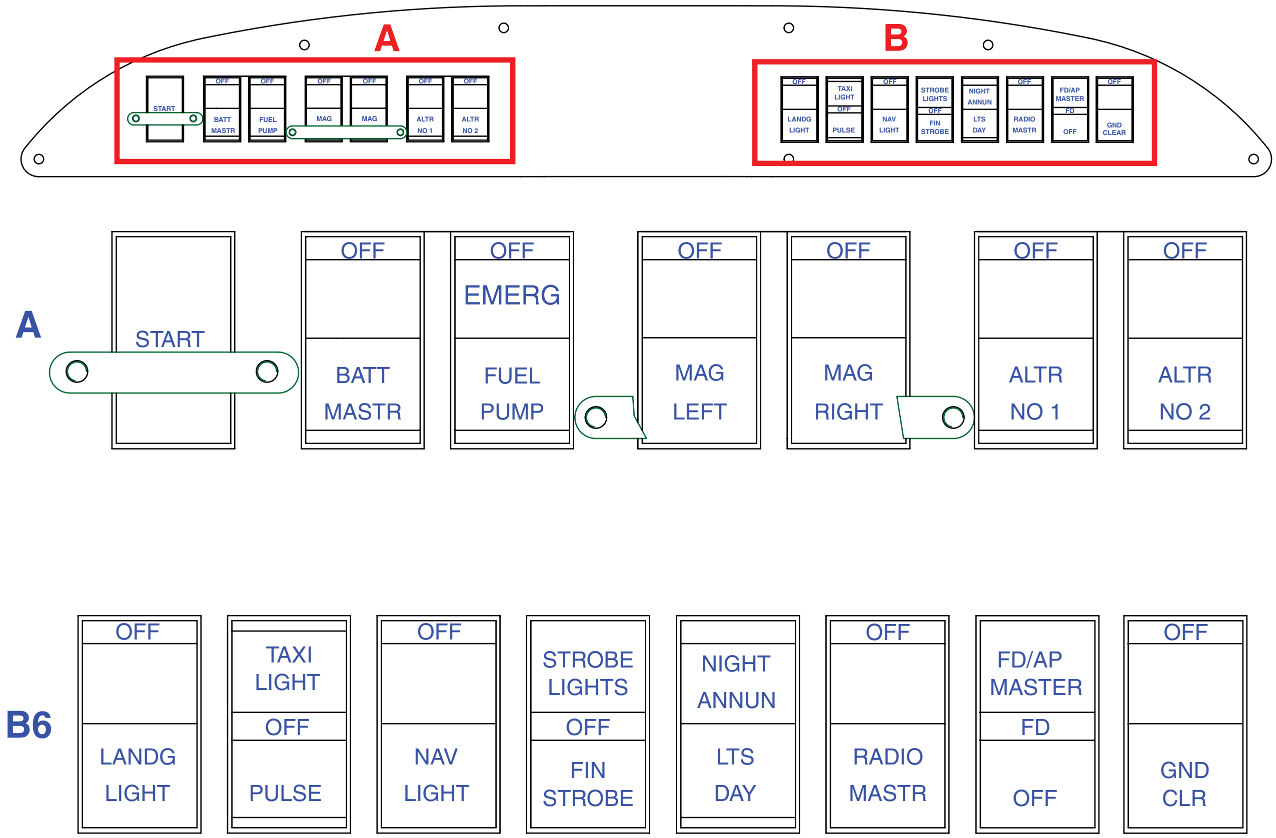
NO SMOKING

(4636375-4636445 Only)

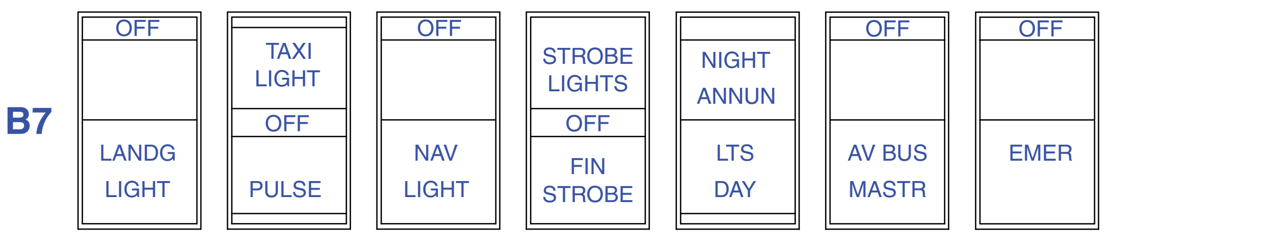
Placards and Markings - Overhead Switch Panel
Figure 17

Effectivity
4636375 and up

105818 D



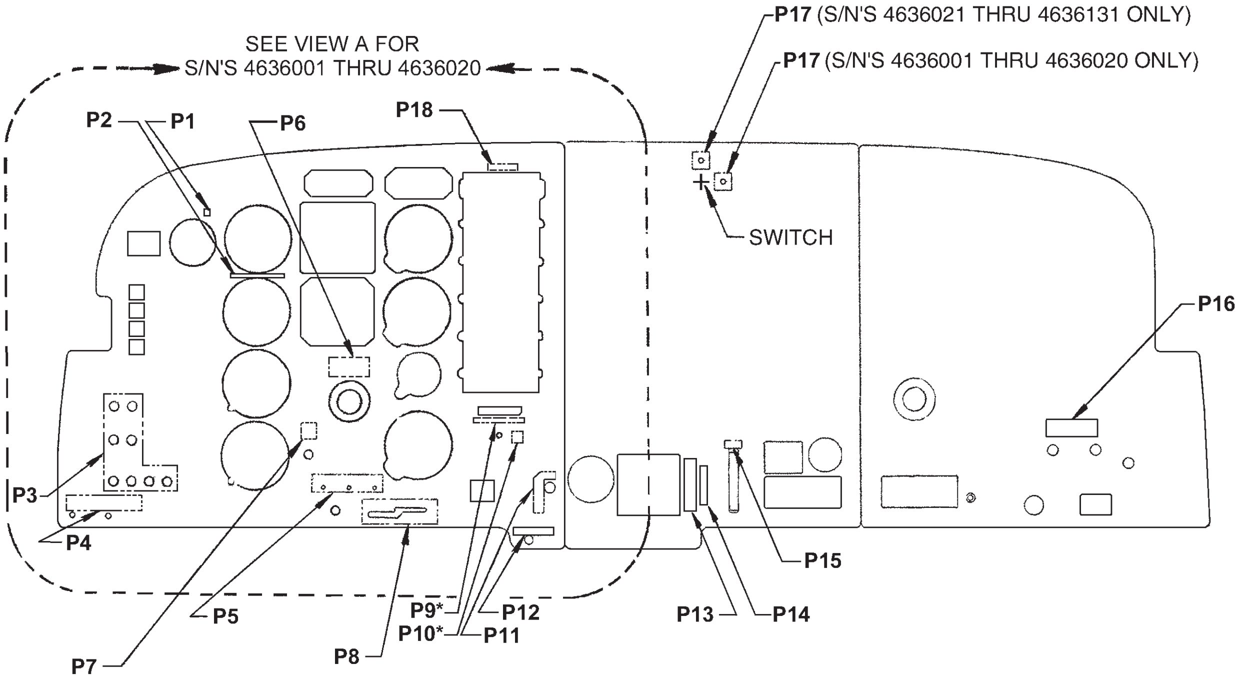
(S/N's 4692001 and up - with Avidyne)



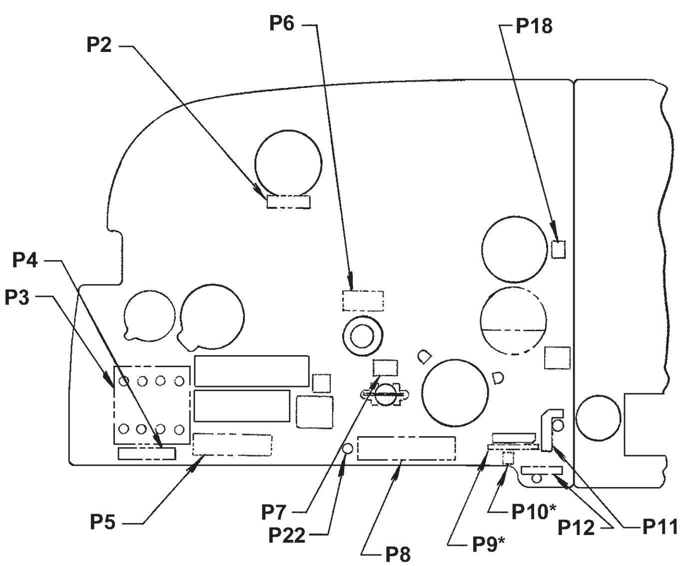
(S/N's 4692134 and up - with G1000)

Effectivity
4692001 and up

Placards and Markings - Overhead Switch Panel
Figure 18



NOTE: Item numbers refer to Decal, Placard, and Transfer detail views shown in later Figures.



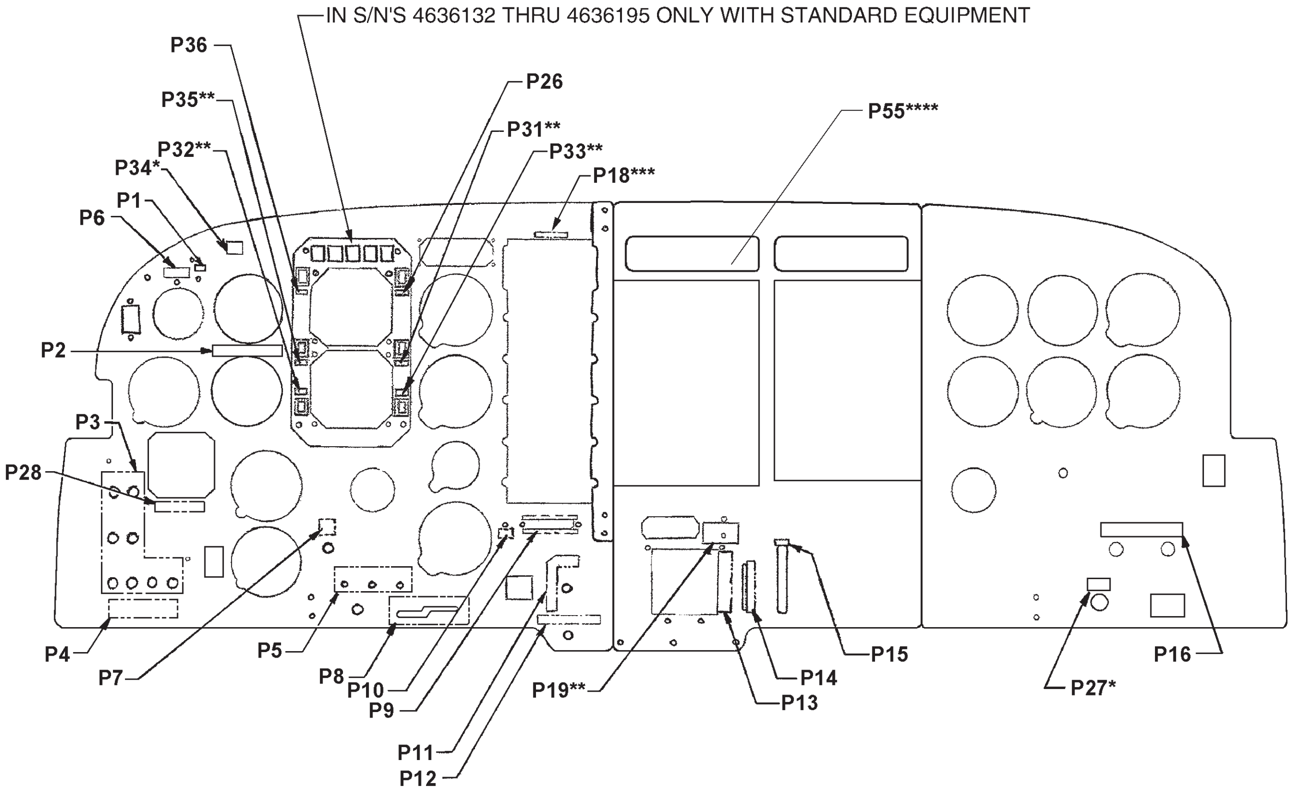
VIEW A
(S/N'S 4636001 THRU 4636020 ONLY)

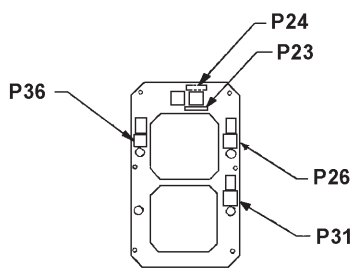
* NOT INSTALLED WITH
SIX (6) CYLINDER CHT GAUGE OPTION.

89502 AZ

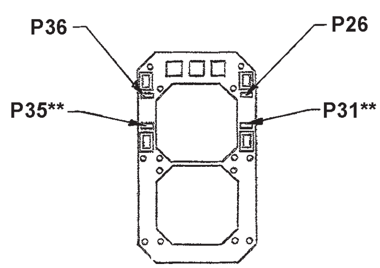
Placards and Markings - Instrument Panel - Overview
Figure 19

Effectivity
4636001 thru 4636374





S/N'S 4636248 THRU 4636313,
LESS 4636299



S/N'S 4636132 THRU 4636195,
WITH EFIS 40 OPTION ONLY

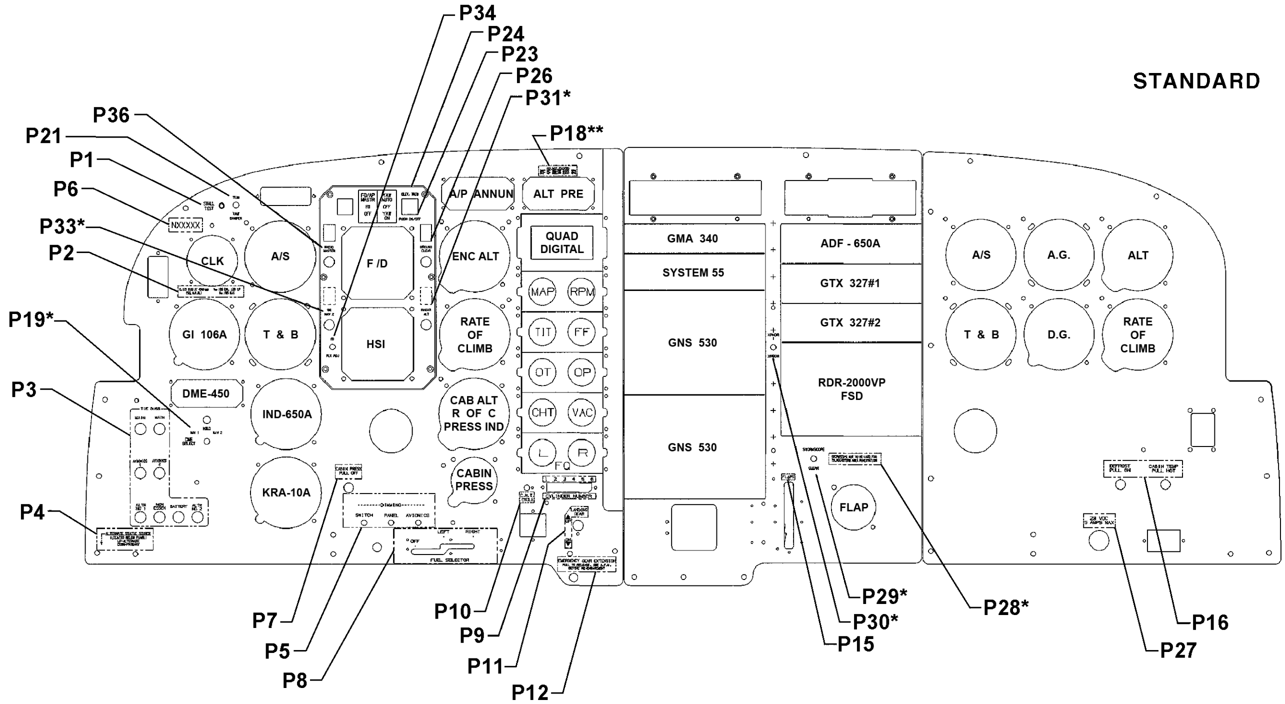
NOTE: Item numbers refer to Decal, Placard, and Transfer detail views shown in later Figures.

* S/N'S 4636248 THRU 4636313, LESS 4636299, ONLY.
** INSTALLED WITH OPTIONAL EQUIPMENT.
*** INSTALLED WITH TWO (2) BLADE PROPELLER OPTION ONLY.
**** INSTALLED WITH PIPER KIT NO. 88549-001 ONLY.

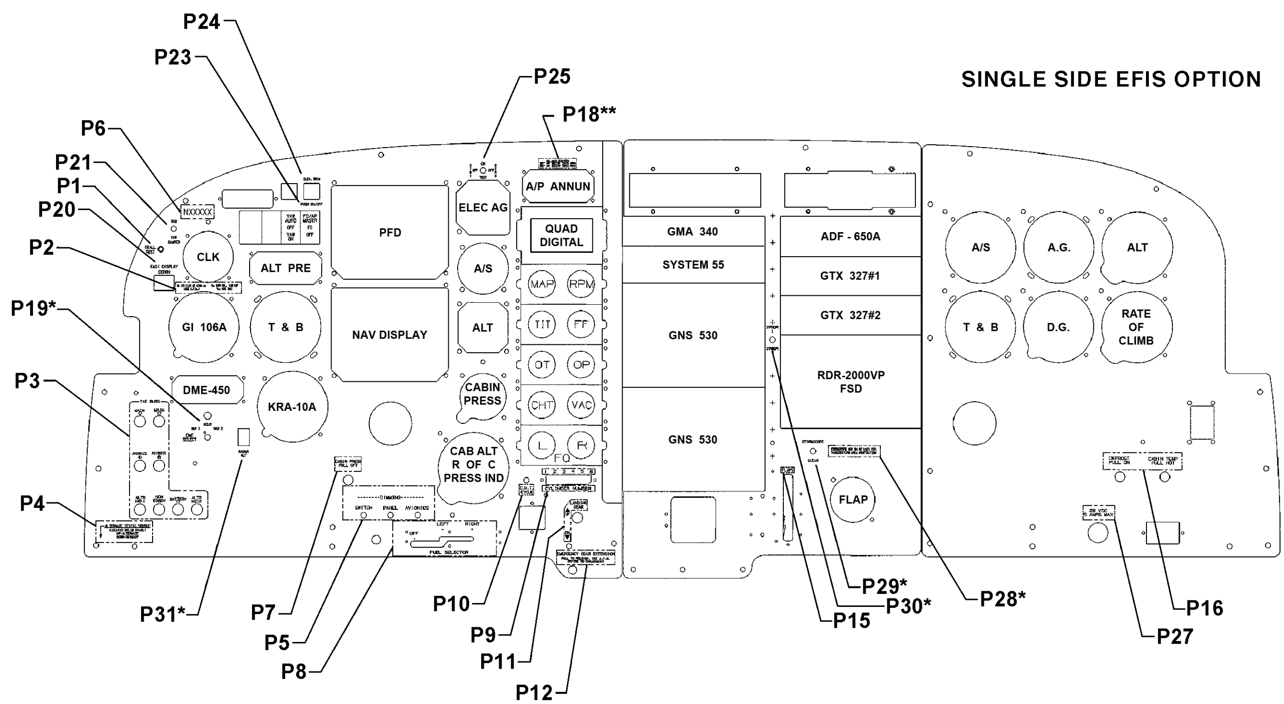
Effectivity
4636132 thru 4636313,
less 4636299

Placards and Markings - Instrument Panel - Overview
Figure 20

STANDARD



SINGLE SIDE EFIS OPTION

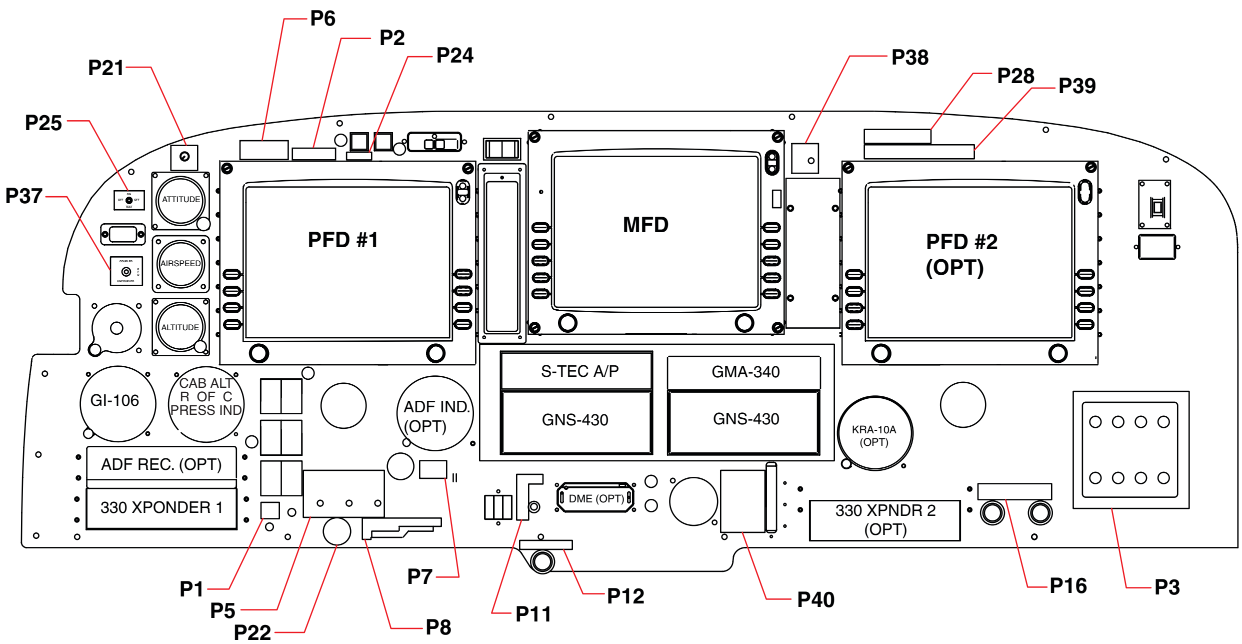


* INSTALLED WITH OPTIONAL EQUIPMENT.
** INSTALLED WITH TWO (2) BLADE PROPELLER OPTION ONLY.

NOTE: Item numbers refer to Decal, Placard, and Transfer detail views shown in later Figures.

Placards and Markings - Instrument Panel - Overview
Figure 21

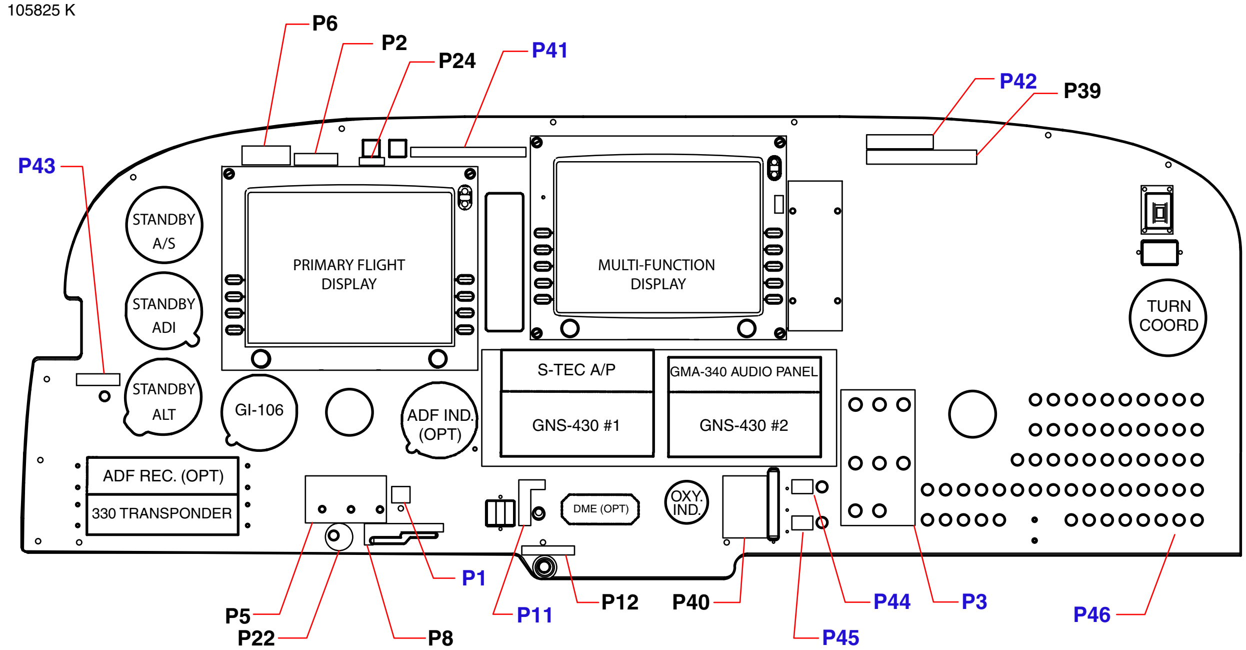
Effectivity
4636299,
4636314 thru 4636374



S/N's 4636375 and up, with Avidyne

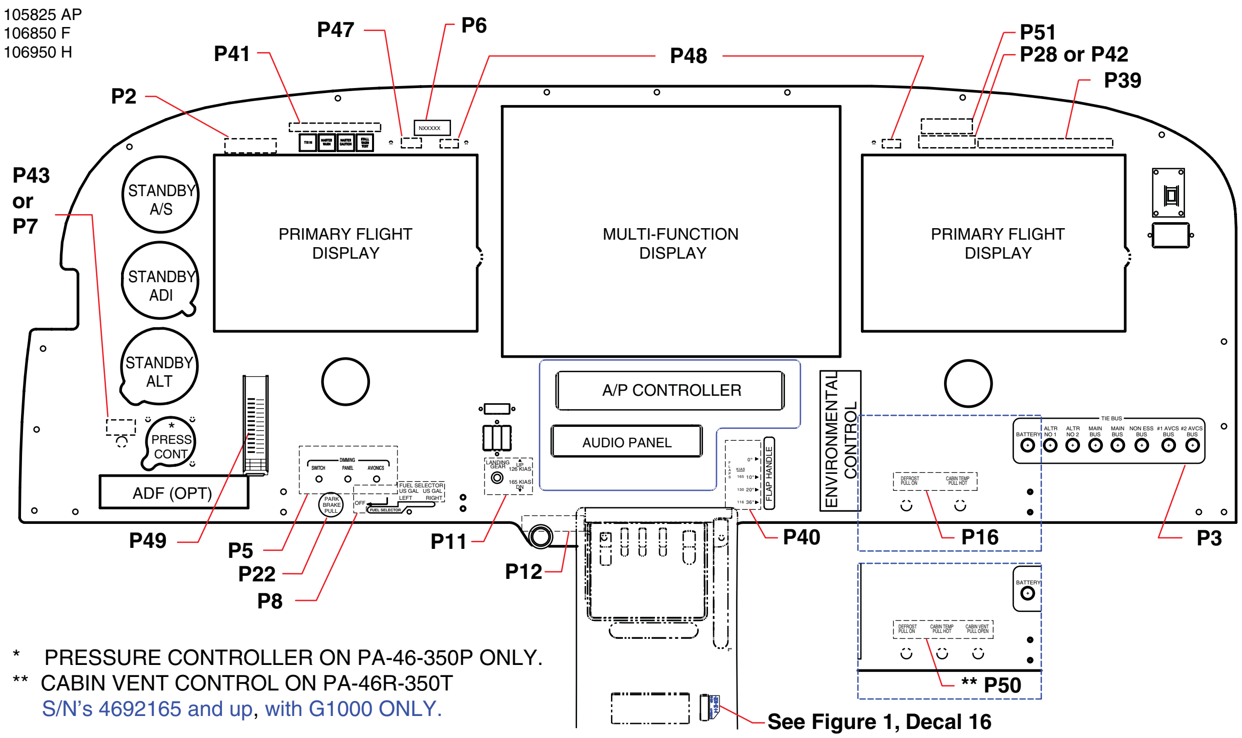
NOTE: Item numbers refer to Decal, Placard, and Transfer detail views shown in later Figures.

105825 K



S/N's 4692001 and up, with Avidyne

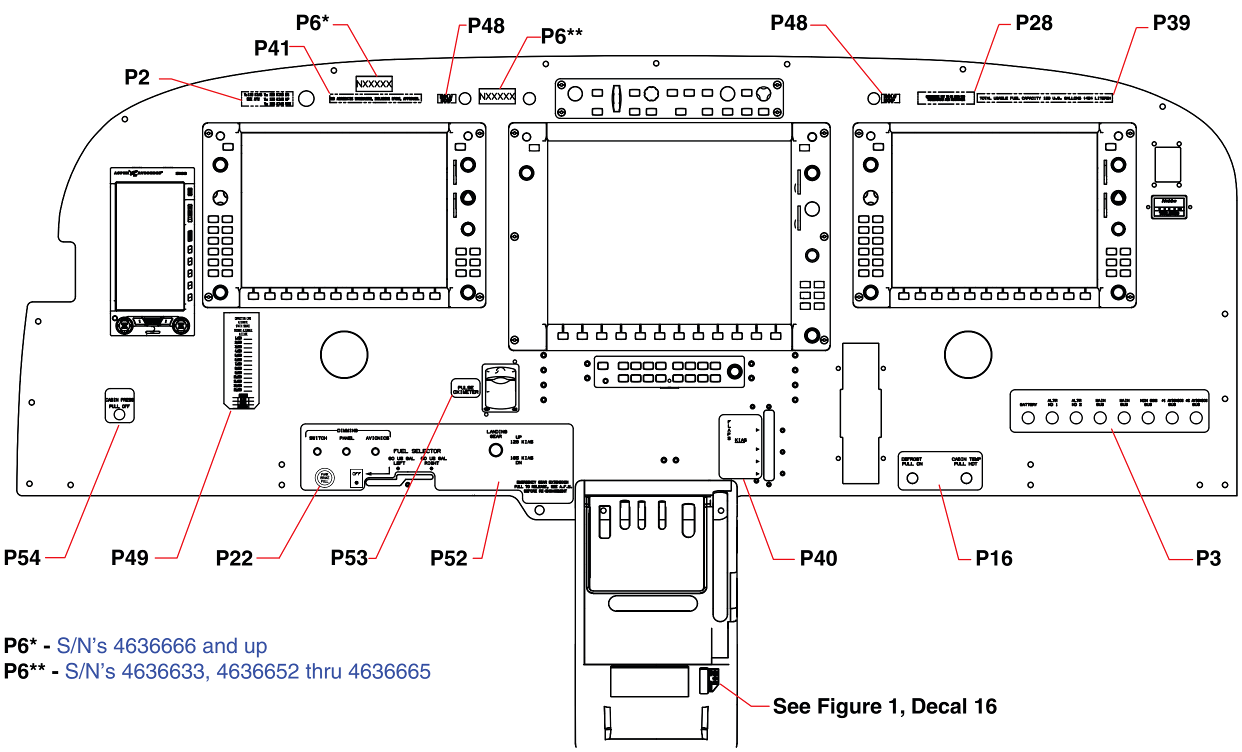
Placards and Markings - Instrument Panel - Overview
Figure 22



* PRESSURE CONTROLLER ON PA-46-350P ONLY.
** CABIN VENT CONTROL ON PA-46R-350T
S/N's 4692165 and up, with G1000 ONLY.

S/N's 4636460, 4636463–4636651 less 4636481 and 4636633; and S/N's 4692134 and up less 4692141, 4692149, and 4692153

NOTE: Item numbers refer to Decal, Placard, and Transfer detail views shown in later Figures.



P6* - S/N's 4636666 and up
P6** - S/N's 4636633, 4636652 thru 4636665

S/N's 4636633, 4636652 and up

Placards and Markings - Instrument Panel - Overview
Figure 23

STALL
TEST

Placard P1
(84967-100)

V_A 133 KIAS AT 4300 LBS. (SEE AFM) V_{LO} 165 DN, 126 UP V_{LE} 195 MAX

V_A 133 KIAS AT 4340 LBS. (SEE AFM) V_{LO} 165 DN, 126 UP V_{LE} 195 MAX

V_A 133 KIAS @ . 4340 LBS V_{LE} 195 KIAS MAX SEE AFM

V_O 118 KIAS SEE AFM V_{LO} 165 KIAS DN V_{LO} 126 KIAS UP V_{LE} 195 KIAS MAX

Decal P2
(See appropriate POH for specific limitations.)

(84967-093 - S/N's 4636001 thru 4636020,
84967-209 - S/N's 4636021 thru 4636195,
84967-258 - S/N's 4636196 thru 4636374,

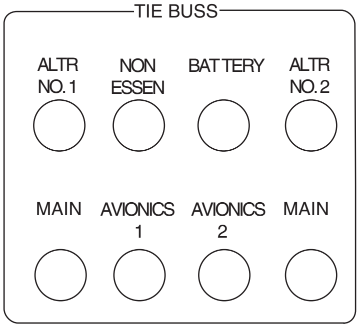
84967-336 - S/N's 4636375 and up with Avidyne,
84967-336 - S/N's 4692001 and up with Avidyne,

84967-347 - S/N'S 4636460, 4636463 and up with G1000,
84967-347 - S/N'S 4692134 and up with G1000.)

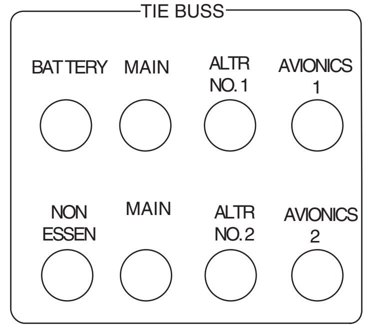
NOTE: Part numbers shown for reference only. See parts catalog to order.

Placards and Markings - Instrument Panel - Detail
Figure 24

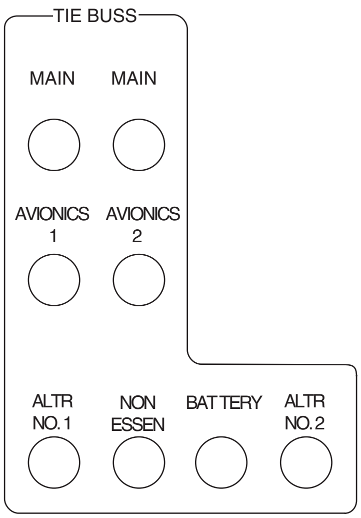
113028



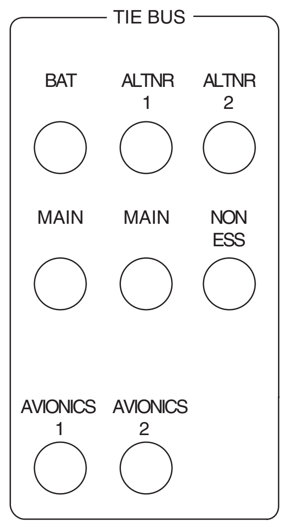
S/N'S 4636001 thru 4636020 only
(84967-091)



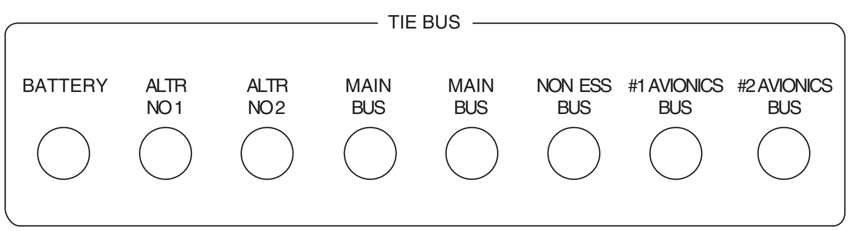
S/N'S 4636375–4636459, 4636161–4636462, 4636481
(84967-337)



S/N'S 4636021 thru 4636374 only
(84967-186 - S/N's 4636021 thru 4636131,
84967-228 - S/N's 4636132 thru 4636374.)



S/N'S 4692001–4692133, 4692141, 4692149, 4692153
(84967-341)



S/N'S 4636460, 4636463–4636651, less 4636481 and 4636633
S/N'S 4692134 and up, less 4692141, 4692149 and 4692153
(84967-346) - Decal
S/N'S 4636633, 4636652 and Up
(107923-101) - EL Placard

P3

Placards and Markings - Instrument Panel - Detail
Figure 25

NOTE: Part numbers shown for reference only. See parts catalog to order.

ALTERNATE STATIC SOURCE
(LOCATED BELOW PANEL)
UP–ALTERNATE
DOWN–PRIMARY

Decal P4
(84967-022)
S/N's 4636001 thru 4636374 only
See Figure 9, Placard #21, for
S/N's 4636375 and up.

DIMMING
SWITCH PANEL AVIONICS

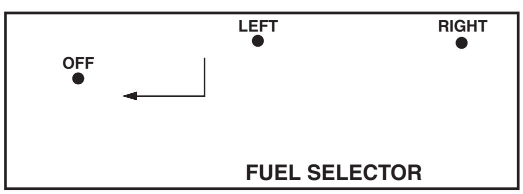
Decal P5
(84967-101)

N XXXXX

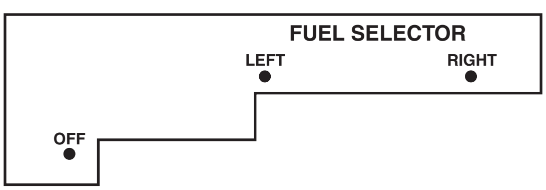
Plate P6
(19099-000)

CABIN PRESS
PULL OFF

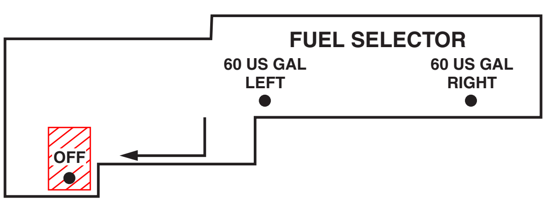
Decal P7
(84967-107)
S/N's 4636001 and up
EL Placard P7
S/N's 4636633, 4636652 and up
107923-014



S/N'S 4636001 thru 4636374 original equipment
(84967-024)



S/N'S 4636375 and up w/ AVIDYNE,
S/N'S 4692001 and up w/ AVIDYNE;
Service replacement for S/N'S 4636001 thru 4636374
(84967-024)



S/N'S 4636460, 4636463 and up w/ G1000,
S/N'S 4692134 and up w/ G1000
(107923-011)

EL Placard P8

Placards and Markings - Instrument Panel - Detail
Figure 26

1 2 3 4 5 6
CYLINDER NUMBER

Decal P9
(84967-109)
(In S/N's 4636132 thru 4636374, cut decal in half and install above and below indicator.)

C.H.T.
CYCLE

Decal P10
(84967-110)

LANDING
GEAR
UP
DOWN

(84967-111)
S/N'S 4636001 thru 4636374

LANDING
GEAR
UP
126 KIAS
165 KIAS
DN

Decal P11
(84967-334)
W/ AVIDYNE installation
S/N'S 4636375 and up
S/N'S 4692001 and up

LANDING
GEAR
UP
126 KIAS
165 KIAS
DN

(84967-345)
W/ G1000 installation
S/N'S 4636460,
4636463 and up
S/N'S 4692134 and up

EMERGENCY GEAR EXTENSION
PULL TO RELEASE. SEE A.F.M.
BEFORE RE-ENGAGEMENT.

Decal P12
(84967-015)

AMP
ALTR
#1
SYST
VOLT
AMP
ALTR
#2

Decal P13
(84967-098)

0
10
20
36

Decal P14
(84967-099)

Placards and Markings - Instrument Panel - Detail
Figure 27

FLAPS

P15
(84967-002 Decal)

DEFROST
PULL ON

CABIN TEMP
PULL HOT

P16
(84967-097 Decal)

(107923-013 EL Placard
S/N's 4636633, 4636652 and Up)

PRESS TO
TEST

P17
(84967-106 Decal)

DO NOT EXCEED

36" MP
BELOW 2400 RPM

32" MP
BELOW 2300 RPM

S/N's 4636001 thru 4636020 only
(84967-195 Decal)

DO NOT EXCEED
36" MP BELOW 2400 RPM
32" MP BELOW 2300 RPM

S/N's 4636021 and up
(84967-208 Decal)

P18

HOLD
NAV 1
NAV 2
DME
SELECT

P19
(84967-244 Decal)

TRIM
YAW
DAMPER

P21
(84967-256 Decal
S/N's 4636299 and 4636314 thru 4636374)
(84967-317 Decal
S/N's 4636375–4636459,
4636461–4636462, 4636481)

PARK
BRAKE
PULL

P22
(35669-081 Placard)

EADI DISPLAY
DOWN

P20
(84967-260 Decal)

PUSH ON / OFF

P23
(84967-216 Decal)

ELEV. TRIM

(84967-215 Decal - S/N's 4636299
and 4636314 thru 4636374)

ELEV. TRIM
PUSH ON/OFF

(84967-316 Decal
S/N's 4636375–4636459,
4636461–4636462, 4636481;
S/N's 4692001–4692133,
4692141, 4692149, 4692153)

P24

STBY
ON
OFF OFF
TEST
GYRO

P25
(84967-267 Decal - when Standby Gyro
Test Switch is required)

GROUND
CLEAR

P26
(84967-241 Decal)

28 VDC
5 AMPS MAX

P27
(84967-259 Decal)

Placards and Markings - Instrument Panel - Detail
Figure 28

STORMSCOPE NOT TO BE USED FOR
THUNDERSTORM AREA PENETRATION

P28
(84967-049 Decal)

STORMSCOPE
CLEAR

P29
(84967-268 Decal)

XPNDR
1
XPNDR
2

P30
(84967-270 Decal)

RADAR
ALT

P31
(84967-240 Decal)

RMI
NAV 2

P33
(84967-243 Decal)

ARGUS
ADF

P35
(84967-238 Decal)

COUPLED
P
F
D
UNCOUPLED

P37
(84967-312 Decal)

A/P
NAV 2

P32
(84967-242 Decal)

FD
PLX ADJ

P34
(84967-247 Decal)

RADIO
MASTER

P36
(84967-239 Decal)

TAS
ON
STBY
TEST

P38
(84967-313 Decal)

TOTAL USABLE FUEL CAPACITY 120 U.S. GALLONS (454 LITERS)

P39
(104349-027 Transfer)

F
L
A
P
S

0° ►

KIAS

165 10° ►

130 20° ►

116 36° ►

P40
(84967-335 Decal)
(107923-012 EL Placard
S/N's 4636633, 4636652 and Up)

NOTE: Part numbers shown for reference only. See parts catalog to order.

Placards and Markings - Instrument Panel - Detail
Figure 29

NO ACROBATIC MANEUVERS, INCLUDING SPINS, APPROVED

P41
(84967-340 Decal)

WEATHER DETECTION SYSTEM NOT TO BE USED FOR
THUNDERSTORM AREA PENETRATION

P42
(84967-342 Decal)

OXYGEN
PULL ON

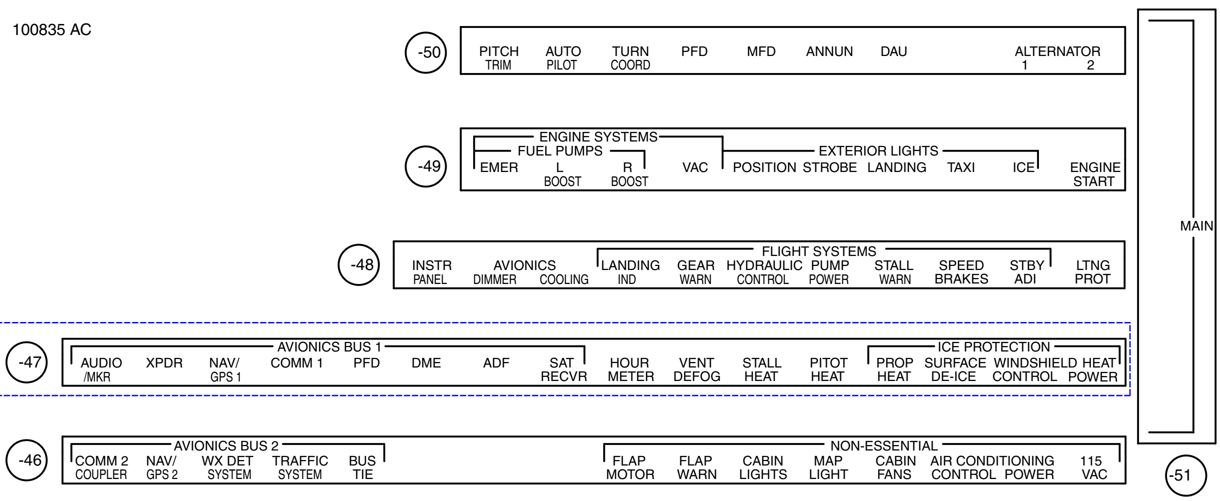
P43
(84967-155 Decal)

DEFROST
PULL ON

P44
(104349-038 Transfer)

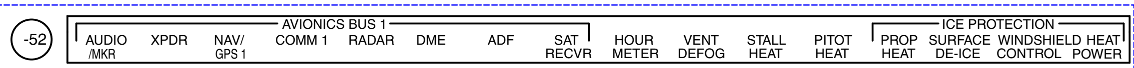
CABIN TEMP
PULL HOT

P45
(104349-039 Transfer)



(100835-046, -047, -048, -049, -050, -051 Placard)

S/N's 4692001 and Up



(100835-052 Placard)

S/N's 4692165 and Up with RADAR OPTION

P46

Placards and Markings - Instrument Panel - Detail
Figure 30

NOTE: Part numbers shown for reference only. See parts catalog to order.

105825 AC
104595 NEW

LTS/GEAR
ANNUN TEST

Decal P47
(84967-349)

DISPLAY
BACKUP

Decal P48
(84967-350)

CORRECTION CARD
ALTERNATE
STATIC SOURCE
PRIMARY ALTERNATE
ALTITUDE

1,000 ____
2,000 ____
3,000 ____
4,000 ____
5,000 ____
6,000 ____
7,000 ____
8,000 ____
9,000 ____
10,000 ____
11,000 ____
12,000 ____
13,000 ____

AIRSPEED

	CLIMB	
	CRUISE	
	APPROACH	

104595-002

Card P49
(104595-002)

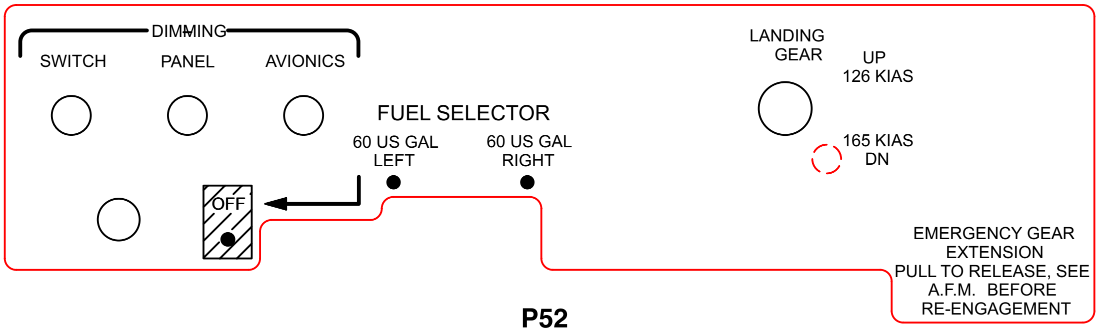
DEFROST PULL ON	CABIN TEMP PULL HOT	CABIN VENT PULL OPEN

Decal P50
(84967-352)

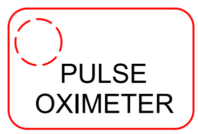
WARNING - THIS AIRCRAFT IS
NOT APPROVED FOR
FLIGHT IN ICING CONDITIONS

Placard P51
(84967-056)
S/N'S 4692134 and up when no ice protection is installed

Placards and Markings - Instrument Panel - Detail
Figure 31



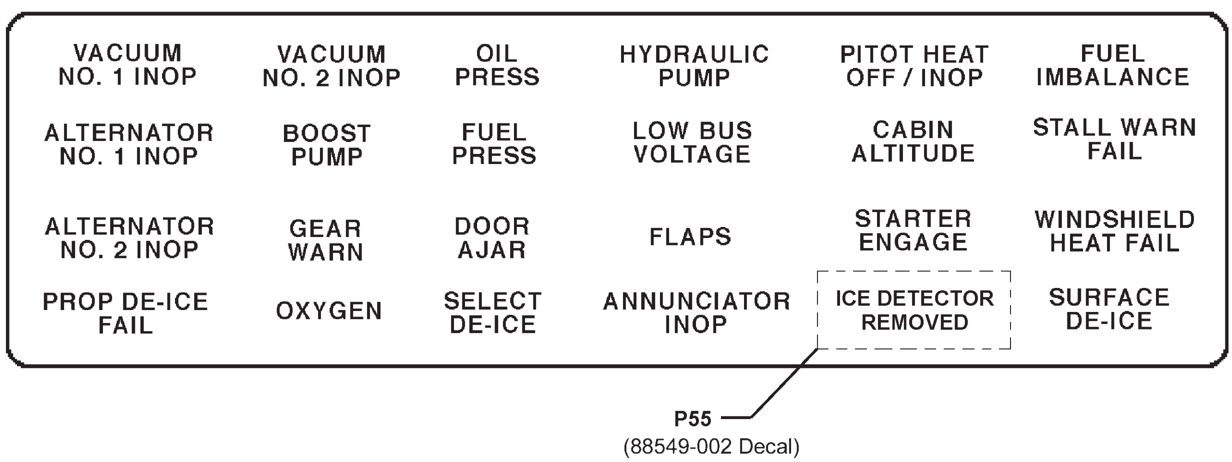
P52
(107923-011 EL Placard
S/N's 4636633, 4636652 and Up)



P53
(107923-015 EL Placard
S/N's 4636633, 4636652 and Up)

P54
(107923-014 EL Placard
S/N's 4636633, 4636652 and Up)

Placards and Markings - Instrument Panel - Detail
Figure 32



NOTE: Part numbers shown for reference only. See parts catalog to order.

Placards and Markings - Instrument Panel - Detail
Figure 33

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CHAPTER

12

SERVICING

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CHAPTER 12

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
12-List of Effective Pages	1	Mar 25/21			
	2	Sep 15/09			
12-Table of Contents	1	Mar 25/21			
	2	Mar 25/21			
12-00-00	1	Nov 30/17			
	2	Nov 30/17			
	3	Nov 30/17			
	4	Nov 30/17			
	5	Nov 30/17			
	6	Nov 30/17			
12-10-00	1	Sep 15/09			
	2	Jan 15/13			
	3	Sep 15/09			
	4	Jan 15/13			
	5	Jan 15/13			
	6	Jan 15/13			
	7	Jan 15/13			
	8	Mar 25/21			
	9	Mar 25/21			
	10	Mar 25/21			
	11	Mar 25/21			
	12	Mar 25/21			
12-20-00	1	Sep 15/09			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Sep 15/09			
	5	Nov 30/17			
	6	Feb 15/14			
	7	Mar 25/21			
	8	Jul 1/10			
	9	Dec 15/14			
	10	Mar 25/21			
	11	Mar 25/21			
	12	Jul 1/10			
	13	Mar 25/21			
	14	Feb 21/20			

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CHAPTER 12 - SERVICING

TABLE OF CONTENTS

CHAPTER 12 - SERVICING

TABLE OF CONTENTS (continued)

GENERAL

This chapter covers all routine servicing of airplane, scheduled and non-scheduled, including replenishment of fuel, oil, hydraulic fluid, oxygen, tire pressure, lubrication requirements, servicing of oleo struts with air and oil, etc. Pay special attention to all WARNINGS or CAUTIONS.

1. Aircraft Finish Care

WARNING: DO NOT USE GASOLINE, KEROSENE, ALCOHOL, BENZENE, CARBON TETRACHLORIDE, THINNER, ACETONE OR WINDOW CLEANING SPRAYS TO CLEAN AIRPLANE.

The entire airplane is carefully finished inside and out to assure maximum service life. The external surfaces are coated with durable polyurethane enamel.

A. Axalta Imron 6000 Paint System

CAUTION: FAILURE TO OBSERVE THE PROPER “FINISH CARE” GUIDELINES MAY RESULT IN DAMAGE OR LOSS OF SHINE OF THE AIRCRAFT PAINT. IMPROPER CARE MAY ALSO VOID THE WARRANTY REGARDING THE AIRCRAFT FINISH.

Piper aircraft delivered in 1999 and later (i.e. - Mirage S/N’s 4636187 and up and Matrix), use the new Axalta Imron 6000 paint system. The guidelines outlined below must be followed to prevent damage to the finish and ensure long paint life.

(1) For the first 30 days after painting:
   (a) Hand wash the aircraft often. Use fresh water only.
   (b) Avoid parking under trees or places where birds roost. If sap, bird droppings, or insect remains are discovered, rinse them off immediately. (Sap, bird droppings, or insect remains will damage the paint during this period.)
(2) For the first 120 days after painting:
   (a) To remove heavy soil, use mild liquid soap. Never use detergent.
   (b) DO NOT WAX THE AIRCRAFT WITHIN 120 DAYS OF PAINTING!
(3) For long term paint finish protection:
   (a) Park in a sheltered area whenever possible.
   (b) Never use a scraper to remove ice or snow from painted surfaces.
   (c) Never let avgas, oil, or hydraulic fluid stand on painted surfaces. (This will permanently damage the finish.)
   (d) Never wash the aircraft in the hot sun.
   (e) Never wipe the finish with a dry cloth, always use fresh water.
   (f) Avoid abrasive cleaners, chemicals, abrasive wax, or brushes.
   (g) Have paint nicks or scratches touched up as soon as possible to maintain the aircraft’s corrosion protection.

To summarize, Piper aircraft using the new Axalta paint system need special attention in the early days of ownership.

B. Cleaning

CAUTION: IF PAINT IS LESS THAN SIX MONTHS OLD, SEE "AXALTA IMRON 6000 PAINT SYSTEM," ABOVE.

CAUTION: DO NOT DIRECT ANY STREAM OF WATER OR CLEANING SOLUTIONS AT THE OPENINGS IN THE PITOT HEAD, STATIC PORTS, ALTERNATE STATIC PORTS OR FUSELAGE BELLY DRAINS.

The airplane should be washed with a mild soap and water solution. Harsh abrasives or alkaline soaps or detergents could scratch painted or plastic surfaces or corrode metal. Cover areas where a cleaning solution could cause damage. To wash the airplane use the following procedure:

(1) Flush away loose dirt with water.
(2) Apply cleaning solution with a soft cloth, a sponge or a soft brush.
(3) To remove exhaust stains, allow the solution to remain on the surface longer.
(4) To remove stubborn oil and grease stains, use a soft cloth dampened with naphtha.
(5) Rinse all surfaces thoroughly.
(6) Any good automotive wax may be used to protect and preserve painted surfaces. Soft cleaning cloths or a chamois should be used to prevent scratches when cleaning or polishing. A heavier coat of wax on leading surfaces will reduce the abrasion problems in these areas. Refer to surface de-ice cleaning procedures.

2. Cleaning

A. Exterior Surfaces

See ""Aircraft Finish Care" on page 12001" and "Windshield and Windows" below.

B. Engine Compartment

Before cleaning the engine compartment:

(1) Place a pan under the engine to catch waste.

CAUTION: DO NOT SPRAY SOLVENT INTO THE ALTERNATOR(S), STARTER, AIR INTAKE, AND ALTERNATE AIR INLETS.

(2) With the engine cowling removed, spray or brush the engine with solvent or a mixture of solvent and degreaser, as desired. It may be necessary to brush areas that were sprayed where heavy grease and dirt deposits have collected in order to clean them.
(3) Allow the solvent to remain on the engine from five to ten minutes, then rinse the engine clean with additional solvent and allow to dry.

CAUTION: DO NOT OPERATE THE ENGINE UNTIL THE EXCESS SOLVENT HAS EVAPORATED OR OTHERWISE BEEN REMOVED.

(4) Lubricate controls, bearing surfaces, etc, per Lubrication Chart, 12-20-00.

C. Windshield and Windows

Except Pilot's Windshield, see below.

WARNING: DO NOT USE GASOLINE, ALCOHOL, BENZENE, CARBON TETRACHLORIDE, THINNER, ACETONE, STRONG SOLVENTS OR WINDOW CLEANING SPRAYS. DO NOT USE PLASTIC CLEANER ON HEATED GLASS WINDSHIELDS.

CAUTION: USE ONLY WATER AND MILD SOAP WHEN CLEANING THE HEATED WINDSHIELD. USE OF ANY OTHER CLEANING AGENT OR MATERIAL MAY CAUSE DISTORTION OR DAMAGE TO WINDSHIELD COATINGS.

(1) Remove dirt, mud and other loose particles from exterior surfaces with clean water.

(2) Wash interior and exterior window surfaces with mild soap and warm water. Use a soft cloth or sponge in a straight back-and-forth motion. Do not rub harshly.

(3) Remove oil and grease with a cloth dampened with Plexiglas Polish and Cleaner, P/N 403D or similar substance conforming to Federal Specification (P-P-560) or kerosene.

(4) Rinse windows thoroughly and dry with soft lint-free cloth.

(5) A superficial scratch or mar in plastic can be removed by polishing out the scratch with jeweler's rouge.

(6) When windows are clean, apply a thin coat of polishing wax. Rub lightly with a soft cloth. Do not apply wax to heated windshields with electrical heating elements.

(7) Apply REPCON repellent or equivalent to windows and windshield to improve visibility during flights through rain. Apply only according to manufacturer's instructions. (Refer to 91-10-00, Consumable Materials.)

D. Pilot's Windshield

WARNING: DO NOT USE GASOLINE, ALCOHOL, BENZENE, CARBON TETRACHLORIDE, THINNER, ACETONE, STRONG SOLVENTS OR WINDOW CLEANING SPRAYS. DO NOT USE PLASTIC CLEANER ON HEATED GLASS WINDSHIELDS.

CAUTION: USE ONLY WATER AND MILD SOAP WHEN CLEANING THE HEATED WINDSHIELD. USE OF ANY OTHER CLEANING AGENT OR MATERIAL MAY CAUSE DISTORTION OR DAMAGE TO WINDSHIELD COATINGS.

The surface of the pilot's windshield should be cleaned in a manner that will protect the special water-repellent coating.

(1) Flush with clean water to remove excessive amounts of dirt and other substances. Adhered particles should be dislodged using fingers or fingernails.

(2) Use only clean materials such as a soft cloth or clean sponge or soft paper towel (such as Kaydry® Wipers). Wash with a 50/50 solution of isopropanol and water. Alternative cleaning solutions that can be used if isopropanol is not available include:

   (a) 50/50 solution of rubbing alcohol and water.

   (b) Mild liquid detergents (such as Ivory Dishwashing Liquid or Joy® Dishwashing Liquid mixed 1/4 oz. per gallon of water).

   (c) Windex® Glass Cleaner

   NOTE: Do not use any abrasive materials (such as pumice or any strong acids or bases as these materials may damage the coating.

(3) Flush thoroughly with clean water and dry. Wipe dry with strokes in one direction using only clean materials such as a damp soft cloth, damp sponge, or soft paper towel (such as Kaydry® Wipers).

NOTE: Do not apply any polish or wax to the glass surface.

E. Landing Gear

(1) Struts and Torque Links

Before cleaning landing gear struts and torque links, place plastic cover or similar material over wheel and brake assembly.

(a) Place can under gear to catch waste.

(b) Spray (low pressure only) or brush gear area, as required, with solvent or mixture of solvent and degreaser.

(c) Allow solvent to remain on gear for 5 to 10 minutes. Rinse gear with additional solvent and allow to dry.

(d) Remove cover from wheel and remove catch-can.

(e) Lubricate gear per Lubrication Chart, 12-20-00.

(2) Wheels and Brakes

CAUTION: DO NOT USE HIGH PRESSURE SPRAY WASH EQUIPMENT. ITS USE CAN INJECT SOAP SOLUTION AND WATER INTO THE WHEEL BEARINGS AND OTHER INTERNAL CAVITIES RESULTING IN CORROSION AND REDUCED SERVICE LIFE.

(a) Hand wash wheels and brakes with a mild soap and water solution.

(b) Rinse with low pressure spray.

(c) Lubricate gear per Lubrication Chart, 12-20-00.

F. Surface Deice Equipment

The deice boots should be cleaned when the aircraft is washed using a mild soap and water solution.

In cold weather, wash the boots with the airplane inside a warm hangar if possible. If the cleaning is to be done outdoors, heat the soap and water solution before taking it out to the airplane. If difficulty is encountered with the water freezing on boots, direct a flow of warm air along the region being cleaned, using a portable type ground heater.

As an alternate cleaning solvent, use benzol or non-leaded gasoline. Moisten the cleaning cloth in the solvent, scrub lightly and then with a clean dry cloth, wipe dry so that the cleaner does not have time to soak into the rubber.

CAUTION: PETROLEUM PRODUCTS SUCH AS THESE ARE INJURIOUS TO RUBBER AND THEREFORE SHOULD BE USED SPARINGLY IF AT ALL.

When deice boots are clean, a coating of B.F. Goodrich Icex should be applied. Icex is compounded to lower the strength of adhesion between ice and rubber surface of the deice boots.

G. Interior

NOTE: Test the cleaner on an inconspicuous portion of the fabric to test for discoloration. Also avoid soaking or harsh rubbing.

(1) Vinyl interior surfaces may be cleaned with a damp cloth and mild soap and water solution.

(2) Leather may be cleaned with a mild hand soap and water solution or with a saddle soap. Follow the precautions which apply to the cleaning of any fine leather product. Avoid saturation and never use detergents or harsh cleaning solutions on leather.

CAUTION: USE OF COMMON HOUSEHOLD CLEANERS AND POLISHES ON WOOD LAMINATED SURFACES COULD BE VERY HARMFUL.

(3) Wood laminated surfaces should be maintained using only a high grade furniture wax.

(4) The optional executive table is hydro-dipped and should be cleaned only as follows:

(a) Dry: Wipe surface with a Micro Fiber cloth to remove dust or debris.

(b) Wet: Use a mild soap and water solution or Windex® with a Micro Fiber cloth.

(5) All upholstery fabrics are Scotchguard treated and may be cleaned as follows:

Spilled oily and watery liquids will generally bead up on the fabric and can be blotted away leaving little or no stain. Blot spills up as quickly as possible with an absorbent cloth, tissue or sponge. If the material is a solid or semi-solid, such as butter, remove the excess by gently scraping with a table knife. Often, blotting will remove all traces of stain but if the staining agent is not completely removed by blotting, the following techniques are suggested:

(6) Water-based stains such as ketchup, milk, ice cream, coffee:

(a) Wipe the stain with a cloth wet with water containing a detergent or ammonia (4 fluid ounces of ammonia to one gallon of water). Repeat if necessary.

(b) Oil based stains such as salad dressing, butter or mayonnaise may be removed by either of the following procedures:

(c) Apply “Texize K-2R Spot Remover” by spraying or rubbing into the fabric and let dry. Vacuum off the residual powder. Repeat if necessary or wet a cloth with a solvent-type spot cleaner such as “Energine” or “Renuzit” and wipe or gently rub the stained area. Turn cloth and rewet with solvent often. Repeat until stain disappears.

(d) To remove residual detergent left on the fabric, wipe the entire fabric surface with a cloth dampened with water. The cloth should be rinsed in clean water several times. This procedure will ensure that the treatment will continue to function.

H. Carpets

WARNING: SOLVENT CLEANERS REQUIRE ADEQUATE VENTILATION.

Use a small whisk broom or vacuum cleaner to remove dirt. For soiled spots, use a nonflammable dry cleaning fluid. Floor carpets may be removed and cleaned like any household carpet.

I. Cleaning and Maintenance of Relief Tube System

CAUTION: THE CORROSIVE AFFECTS OF URINE ON PAINTED AND UNPAINTED SURFACES CANNOT BE UNDERSTATED. CORROSION MAY APPEAR IN SURROUNDING AREAS IF CLEANING IS DEFERRED FOR EVEN ONE DAY!

For airplanes equipped with a relief tube system, the corrosive effects of urine or other liquids poured through the system are extreme and heightened attention to the cleanliness of this system is indicated, both inside and outside the aircraft. In the interior, the funnel tube assembly, rubber hose and surrounding sheet metal must be cleaned at the termination of each flight, if the system has been used. Likewise, attention to the exterior of the aircraft is equally important and exterior surfaces must be cleaned as described below.

(1) Interior

CAUTION: SHOULD SPILLAGE EXTENDING INTO THE FUSELAGE BE EVIDENT, PROMPT MAINTENANCE ACTION IS REQUIRED TO ENSURE URINE SPILLAGE IN THE FUSELAGE STRUCTURE IS NEUTRALIZED. REMOVE PANELS TO ACCESS THE FLOOR STRUCTURE AS REQUIRED.

(a) After each use of the relief tube, the area surrounding the relief tube should be examined for spillage and cleaned according to the procedures in paragraphs E (4) and E (5), above. Clean the area inside the box and access door, funnel, and tube using mild soap and water. After cleaning, flush with clean water to ensure that no soapy residue remains. Dry system thoroughly.

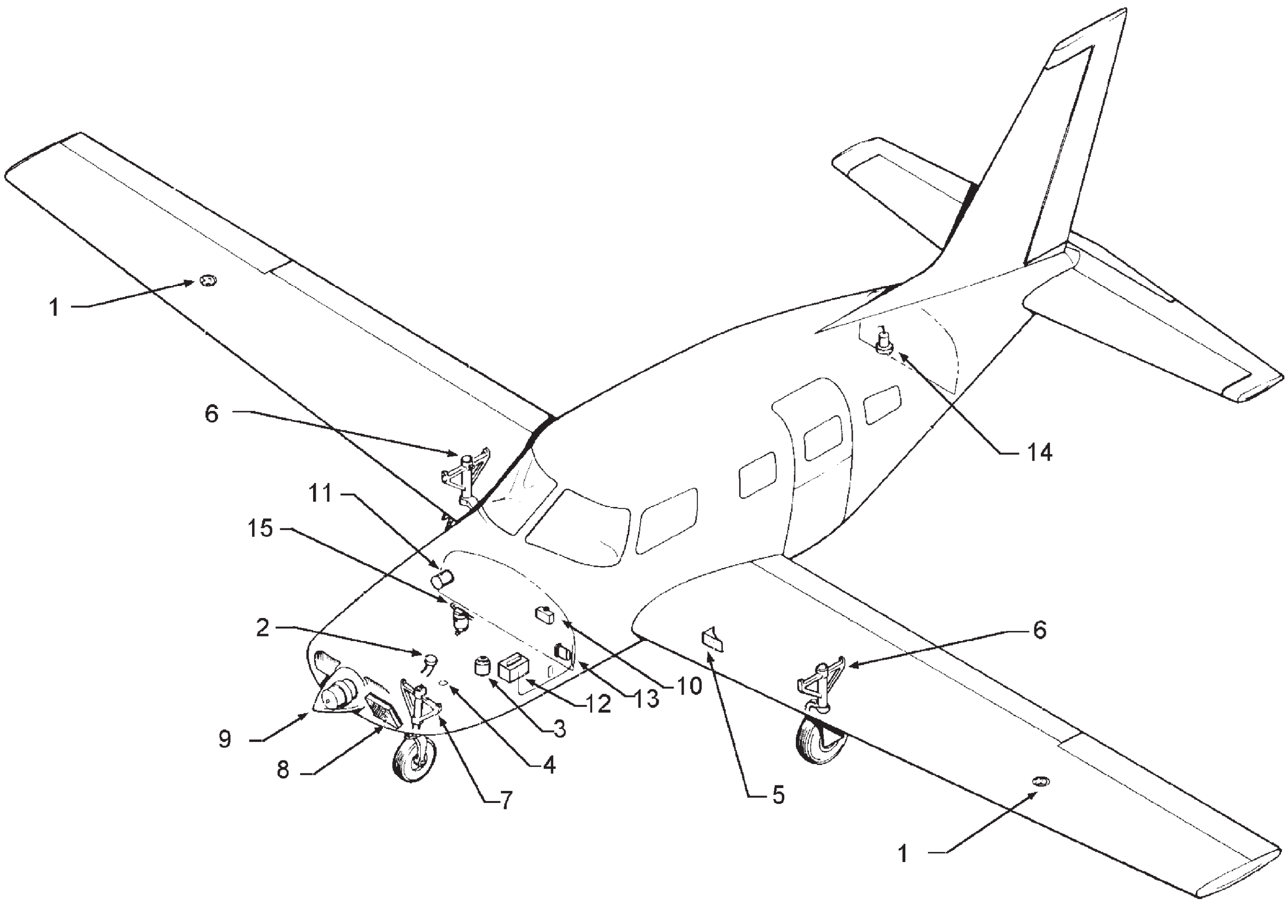
(b) Prepare to flush the relief tube assembly by placing a catch-can underneath the relief tube outlet. Flush tube by pouring a solution of baking soda (10%) and water through the tube, flushing out the entire system. Flush again with at least 1/2 gallon of clean water. Low pressure (LP) shop air may be blown through the relief tube system to dry it.

(2) Exterior

Exterior bottom painted surfaces of the airplane must be cleaned from the firewall to the tip of the tail, including the bottom of the tail surfaces, at termination of each flight during which the relief tube system was used. Cleaning should occur per paragraph C, above, with the exception that after completion of washing, a solution of baking soda (10%) and water should be applied to the entire area and allowed to remain for a few minutes. The area must then be thoroughly rinsed with clean water. Dry the area thoroughly and inspect for paint chipping and/or corrosion. Touch up paint as necessary.

REPLENISHING

1. Service Points



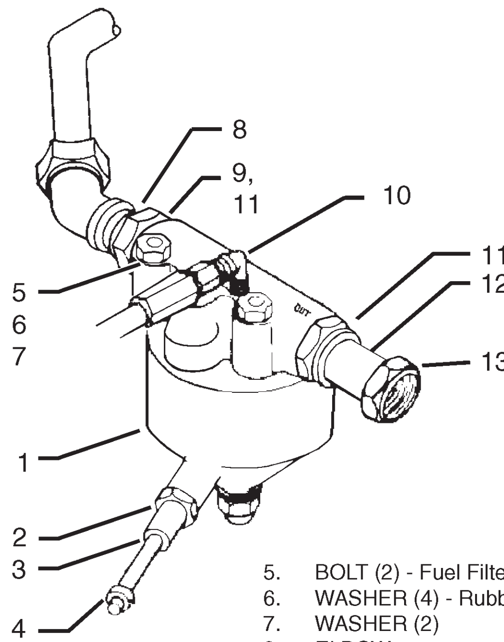
1. FUEL TANK FILLERS
2. ENGINE OIL FILLER
3. ENGINE OIL FILTER
4. ENGINE OIL DRAIN
5. FUEL DRAIN (L/H SHOWN)
6. MLG FILLER VALVE
7. NLG FILLER VALVE
8. INDUCTION AIR FILTER
9. PROPELLER HUB
10. BRAKE RESERVOIR
11. GYRO AIR FILTER
12. BATTERY
13. EXTERNAL POWER RECEPTACLE
14. HYDRAULIC PUMP/RESERVOIR
15. FUEL FILTER/DRAIN

Service Points
Figure 1

2. Fuel System

A. Fuel Filter

At intervals of 50 hours or 90 days, whichever comes first, clean (or replace) the fuel filter, located in the fuel bowl mounted on the lower right side of the forward baggage compartment (refer to 28-20-00, Figure 3), below the floors, and accessible through the fuel filter access plate on the right side of the fuselage. Remove and clean the fuel filter (refer to "Figure 2" and "Figure 2a"), in accordance with the instructions outlined below. Additional fuel system service information may also be found in Chapter 28. Inspection intervals of the various fuel system components may be found in 5-20-00.



1. FUEL FILTER ASSEMBLY
2. O-RING - Fuel Filter Drain
3. VALVE - Fuel Filter Drain
4. BUSHING - Fuel Filter Drain
5. BOLT (2) - Fuel Filter
6. WASHER (4) - Rubber
7. WASHER (2)
8. ELBOW
9. NUT
10. FITTING - ELBOW
11. O-RING - Fuel Filter & Pump
12. UNION - Fuel Filter/Pump
13. NUT

Fuel Filter Assembly - Typical
Figure 2

(1) Removal

(a) Determine that the fuel selector valve is in the OFF position.

(b) Just outboard of the forward baggage compartment, and forward of the right wing, remove the fuel filter access panel.

(c) With the drain cup, drain any fuel still in the filter from the Drain Valve. Pushing in on drain tube opens Drain Valve.

(d) Cut the safety wire and remove nut and washers/gasket from bottom of bowl.

(e) Remove bowl and O-ring from body.

(f) Remove the check and retaining nuts from the stud, and slide the filter element/cartridge assembly down off the stud.

(2) Installation

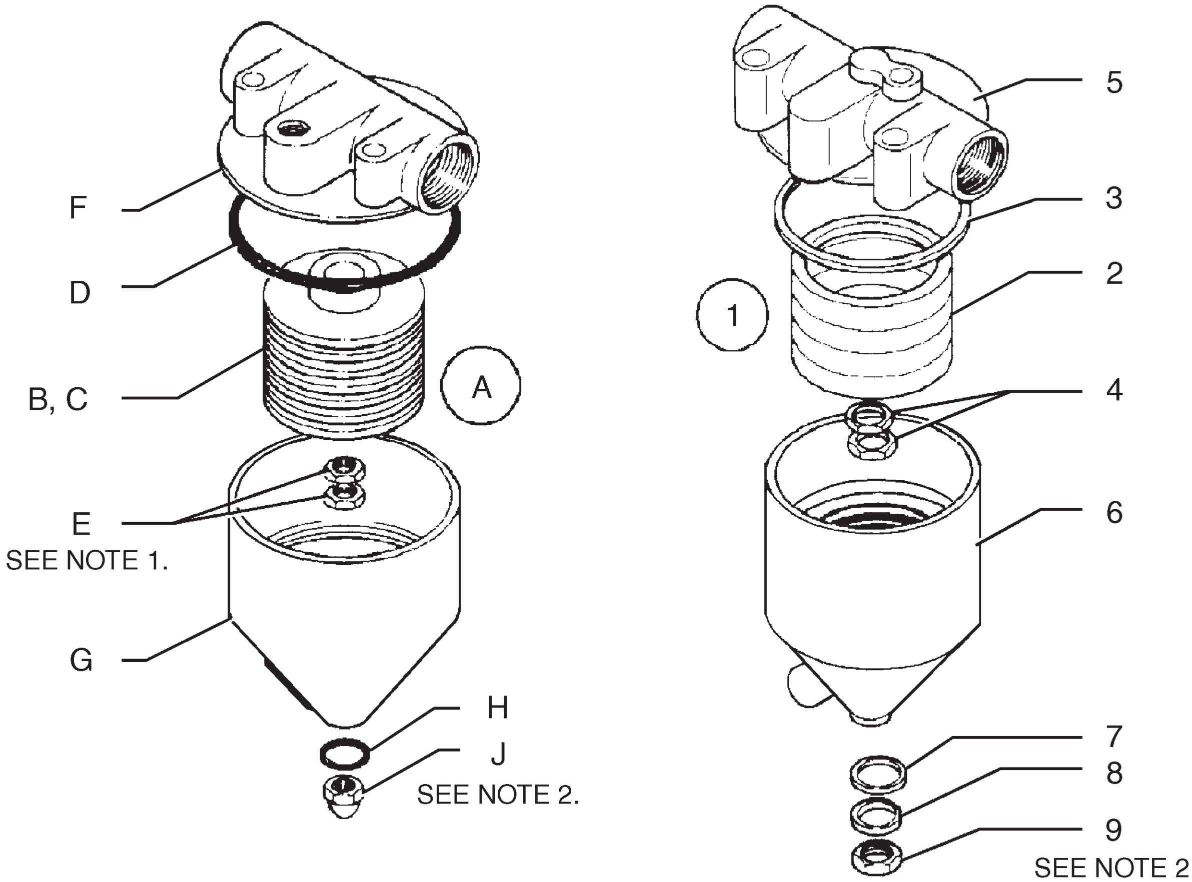
(a) Install filter element/cartridge assembly onto stud

(b) Secure assembly with nuts. Torque as specified in "Figure 2a"

(c) Install new O-ring on top of bowl assembly, place bowl in position on filter body and stud. Determine that the drain valve connection is in line with the access plate inlet. Secure with washers/gasket and nut. Torque as specified in "Figure 2a".

(d) Safety nut appropriately.

(e) Determine that the drain valve on the side of the fuselage is closed and place the fuel selector valve lever in the desired position. Replace access plate on side of fuselage.



NOTES:
1. TORQUE TOP NUT 10 TO 15 LB. IN. HOLDING THIS NUT IN PLACE, TORQUE LOWER NUT 40 TO 60 LB. IN.
2. TORQUE NUT 60 TO 80 LB. IN.

A. FUEL FILTER Assy
   599-180 02W072322
   Air Maze
B. CARTRIDGE Assy - Filter
C. Filter Disc (7)
D. PACKING - O-RING
E. NUT - Hex, Jam
F. HEAD Sub-Assy - Filter
G. CONTAINER - Filter
H. GASKET - Container Nut
J. NUT - Container

1. FUEL FILTER Assy
   561-028 7587376
   Purolator
2. ELEMENT - Filter
3. PACKING - O-RING
4. NUT - Hex
5. HEAD Assy - Fuel Filter
6. CASE Assy - Fuel Filter
7. WASHER - Seal
8. WASHER - Countersunk
9. NUT - Hex, Drilled

Fuel Filter Assemblies
Figure 2a

(3) Cleaning

(a) Clean filter element/cartridge assembly and bowl thoroughly using acetone or other suitable dry type cleaning solvent.

(b) Remove filter from the cleaner and rinse thoroughly in clean hot flowing tap water.

(c) Dry using a light blast of compressed air.

(d) Inspect the filter discs for damage and/or broken screens.Replace screen if any signs of damage or deterioration are evident.

(e) Check condition of bowl seal washer.

(f) Check for corrosion of filter parts.

(g) Discard O-ring gasket and replace it with a new one at reassembly.

B. Filling Fuel Cells

WARNING: DURING FUEL SYSTEM MAINTENANCE PRACTICES, A GROUND WIRE ATTACHED FROM EACH MAIN GEAR GROUNDING PIN TO SEPARATE APPROVED GROUNDING STAKES SHOULD BE USED TO PREVENT UNGROUNDING OF THE AIRCRAFT DUE TO ACCIDENTAL DISCONNECTION OF ONE GROUND WIRE.

Observe all required precautions for handling gasoline. Fill the fuel tanks with the fuel as specified on the placard adjacent to the filler neck. Each wing tank has a capacity of 61 U.S. gallons, for a total capacity of 122 U.S. gallons. Two U.S. gallons are considered unusable fuel.

C. Water Contamination

WARNING: PILOTS, OWNERS, OPERATORS, MAINTENANCE, AND SERVICE PERSONNEL SHOULD ASSUME SOME WATER EXISTS IN THE FUEL SYSTEM.

Water may enter the fuel tank system via any penetration in the wing fuel tank and from moisture condensation inside the tank. Water in the fuel may come out of solution, settle and make its way to a drain location in the form of a blob, pea, or BB-shaped translucent mass found at the bottom of the sampler cup. Water suspended in the fuel may lead to a cloudy or hazy appearance in the sampler cup. Water may have dissolved in the fuel, but conditions have not yet occurred to cause the water to come out of solution and perhaps adhere to the dry tank upper surface or walls (similar to condensation).

See FAA Special Airworthiness Information Bulletin (SAIB) No. CE-12-06 for additional information.

D. Fuel Anti-icing Inhibitors

CAUTION: ASSURE THAT THE ADDITIVE IS DIRECTED INTO THE FLOWING FUEL STREAM. THE ADDITIVE FLOW SHOULD START AFTER AND STOP BEFORE THE FUEL FLOW. DO NOT PERMIT THE CONCENTRATED ADDITIVE TO COME IN CONTACT WITH THE AIRCRAFT PAINTED SURFACES OR THE INTERIOR SURFACES OF THE FUEL TANKS.

CAUTION: SOME FUELS HAVE ANTI-ICING ADDITIVES PREBLENDED IN THE FUEL AT THE REFINERY, SO NO FURTHER BLENDING SHOULD BE PERFORMED.

CAUTION: THIS ADDITIVE SHOULD NOT BE USED AS A REPLACEMENT FOR PREFLIGHT DRAINING OF THE FUEL SYSTEM DRAINS.

The MIL-DTL-85470 Fuel Anti-Icing Inhibitor, primarily is an anti-icing agent, however it also has excellent microbiological sludge-deterrent characteristics. It is very soluble in water, and only slightly soluble in fuel, which mandates that the blending with fuel be accomplished in a precise manner. Such a device permits injection of the agent into a flowing stream of fuel to ensure even disbursement. When blending the additive, the concentration should not be less than 0.10 percent or more than 0.15 percent by volume. Except for the information contained herein and in the Pilot's Operating Handbook, the manufacturer's mixing or blending instructions should be carefully followed.

E. Draining Moisture From Fuel System

CAUTION: WHEN DRAINING ANY AMOUNT OF FUEL, CARE SHOULD BE TAKEN TO ENSURE THAT NO FIRE HAZARD EXISTS BEFORE STARTING THE ENGINE.

The fuel tanks and fuel filter should be drained daily prior to first flight and after refueling to avoid the accumulation of water or sediment. Each fuel header tank is equipped with an individual quick drain located at the lower inboard rear corner of the tank.

The fuel filter with a quick drain valve is located on the lower right side of the forward baggage compartment.

NOTE: There is no single point of drainage that can be used to check for all fuel system contaminants simultaneously.

Drain fuel tanks and filter per the following:

With the airplane in the normal ground attitude and starting at the highest drain location, check all drain locations for contaminants before every flight, whether or not refueling has occurred. Have fuel sample disposal provisions and proper lighting at your disposal to properly check for fuel tank system contamination.

(1) Drain at least one cup of fuel (using a clear sampler cup) from each drain location:
   (a) drain each wing through its individual quick drain located at the lower inboard rear corner of the header tank;
   (b) drain the fuel filter as required to completely flush its contents in each of the fuel selector positions. Drain the fuel filter by opening the quick drain valve on the filter.
(2) Check for water, clarity, cloudiness, haze, proper fuel type/grade (i.e.; 100LL is light blue in tint, jet fuel is clear or yellowish), odor, or other contaminants.
(3) Allow time between fueling and draining. It takes time for any contaminates to settle to sump area prior to draining tanks.
(4) If any contamination is detected in the fuel tank system, thoroughly drain all drain locations again.
(5) If contamination is observed, take further samples until the fuel appears clear, and gently rock the airplane in both the roll and pitch axis to move any additional contaminants to the drain points.
(6) Take repeated samples from all drain locations until all contamination has been removed.
(7) If contaminants are still present, do not fly the airplane. Have qualified maintenance personnel drain and purge the fuel tank system. Remove all evidence of contamination prior to further flight.

F. Draining Fuel System - Complete

Fuel may be drained from the system by opening the valve at the inboard end of each fuel tank. The drain valve requires the drain cup pin to hold the valve open. The remaining fuel in the system may be drained through the filter bowl.

3. Landing Gear

A. Oleo Struts

(PIR-PPS50045, Rev. L.)

WARNING: DO NOT RELEASE AIR BY REMOVING THE STRUT VALVE CORE OR FILLER PLUG. DEPRESS THE VALVE CORE PIN UNTIL STRUT CHAMBER PRESSURE HAS DIMINISHED.

CAUTION: DIRT AND FOREIGN PARTICLES FORM AROUND THE FILLER PLUGS OF THE LANDING GEAR STRUTS, THEREFORE, BEFORE ATTEMPTING TO REMOVE THESE PLUGS, THE TOPS OF THE STRUTS SHOULD BE CLEANED WITH COMPRESSED AIR AND/OR WITH A QUICK DRYING SOLVENT.

Air-oil type oleo struts should be maintained at proper strut tube exposure for best oleo action. Nose gear strut piston tube exposure is 1.7 ± .25 inches. Main gear strut piston tube exposure is 3.5 ± .25 inches - except in S/N's 4636196 and up and S/N's 4692001 and up where 3.4 ± .25 inches is required. Measure with the airplane sitting on a level surface under normal static load. (Empty weight of airplane plus full fuel and oil). If the strut has less tube exposure than prescribed, determine whether it needs air or oil by rocking the airplane. If the oleo strut oscillates with short strokes (approximately one inch) and the airplane settles to its normal position within one or two cycles after the rocking force is removed, the oleo strut requires inflating. Check the valve core and filler plug for air leaks, correct if required, and add air or nitrogen as described in Inflating Oleo Struts. If the oleo strut oscillates with long strokes (approximately three inches) and the airplane continues to oscillate after the rocking force is removed, the oleo struts require fluid. Check the oleo for indications of oil leaks, correct if required and add fluid as described in Filling Oleo Struts. For landing gear and/or oleo strut repair procedures, refer to 32-10-00 and 32-20-00.

(1) Filling Oleo Struts

To fill the nose or main gear oleo strut with fluid (MIL-PRF-5606), whether it be the addition of a small or large amount, proceed as follows:

(a) Raise the airplane on jacks.

(b) Place a pan under the gear to catch spillage.

(c) At the filler plug, relieve air pressure from the strut housing chamber by removing the cap from the air valve and depressing the valve core.

(d) There are two methods by which the strut chamber may be filled. They are as follows:

1 METHOD I.

a Remove the valve core from the filler plug at the top (or side) of the gear strut housing. Allow the filter plug to remain installed.

b Attach one end of a clean plastic hose to the valve stem of the filler plug and submerge the other end in a container of hydraulic fluid.

NOTE: An air-tight connection is necessary between the plastic tube and the valve stem. Without such a connection, a small amount of air will be sucked into the oleo strut during each sequence, resulting in an inordinate amount of air bubbles and a prolonged filling operation.

c Fully compress and extend the strut thus drawing fluid from the fluid container and expelling air from the strut chamber. By watching the fluid pass through the plastic hose, it can be determined when the strut is full and no air is present in the chamber.

d When air bubbles cease to flow through the hose, compress the strut fully and remove the hose from the valve stem.

e With the strut compressed, remove the filler plug to determine that the fluid level is visible up to the bottom of the filler plug hole.

f Reinstall the core in the filler plug and apply thread lubricant (Kopr-Kote or equivilent (MIL-PRF-907E)) to the threads of the filler plug and install the plug in the top of the strut housing. Torque the plug from 350 to 400 inch pounds.

2 METHOD II.

a Remove the filler plug from the top (or side) of the gear strut housing.

b Raise the strut piston tube until it is fully compressed.

c Pour fluid from a clean container through the filler opening until it reaches the bottom of the filler plug hole. (Air pressure type oil container may be helpful.)

d Install the filler plug finger tight and extend and compress the strut two or three times to remove air from the housing.

e Remove the filler plug, raise the strut to full compression and fill with fluid if needed.

f Apply thread lubricant (Kopr-Kote or equivilent (MIL-PRF-907E)) to the threads of the filler plug. Reinstall the filler plug and torque from 350 to 400 inch pounds.

g With the airplane raised, compress and extend the gear strut several times to ascertain that the strut actuates freely. The weight of the gear fork and wheel should extend the strut.

h Clean off overflow of fluid and inflate the strut as described in Inflating Oleo Struts.

i Check that fluid is not leaking around the strut piston at the bottom of the housing.

(2) Inflating Oleo Struts

After making certain that an oleo strut has sufficient fluid, attach a strut pump to the air valve and inflate the oleo strut. The strut should be inflated until the correct inches of piston is exposed with normal static load (empty weight of the airplane plus full fuel and oil) on the gears. Rock the airplane several times to ascertain that the gear settles back to the correct strut position. If a strut pump is not available, the airplane may be raised and line pressure from a high pressure air system used.

NOTE: Moisture free compressed air may be used in lieu of nitrogen gas on a temporary basis; however, the strut should be deflated and serviced with nitrogen as soon as possible.

Lower the airplane and, while rocking it, let air from the valve to bring the strut down to the proper extension. (Before capping the valve, check for valve core leakage.)

If the airplane is on jacks, ensure that the struts are fully extended. Inflate the nose gear strut to 110 ± 11 psi and the main gear struts to 250 ± 25 psi. This is the preferred method.

B. Brake System

The brake system contains a hydraulic fluid reservoir through which the brake system is periodically serviced. Fluid is drawn from the reservoir by the brake master cylinders to maintain the volume of fluid required for maximum braking efficiency. Spongy brake pedal actions is often an indication that the brake fluid reservoir is running low on fluid or air. When repairs to brake system components or bleeding the system are required, refer to instructions in 32-40-00.

(1) Filling Brake System Reservoir

The brake system reservoir must be filled with hydraulic fluid (MIL-PRF-5606) to the level marked on the reservoir. The reservoir, located on the left side of the aft wall of the forward baggage compartment, must be checked at every 50 hour inspection and replenished as necessary. Access to the reservoir is through the door on the upper right portion of the nose section. If fluid level is low, it must be filled with filtered hydraulic fluid (MIL-PRF-5606). No adjustment to the brakes is necessary, but they must be checked periodically per instructions given in 32-40-00.

(2) Draining Brake System

To drain the brake system, disconnect the supply hose from the fitting on the bottom of the cylinder and open the bleeder valve at the top of the cylinder. This will empty the piston chamber. Pump the desired brake pedal to clear the line of fluid. To clean system, flush with denatured alcohol.

NOTE: If brake system is drained, parts replaced, or line connections broken, pressure fill and bleed system per procedure in 32-40-00. Procedure assures air is not trapped in system.

C. Tires

There are several safety benefits to proper tire maintenance, including its effect on ground steering. Maintaining proper tire inflation pressure reduces the size of a tire’s rolling contact patch, which reduces the magnitude of the force associated with ground steering.

According to tire manufacturers, tire pressure should be checked daily, prior to the first flight of the day, when the tires are “cold.” (A cold tire is one that has the same temperature as the ambient environment, and has not been operated for at least three hours.) If an aircraft is flown less than one time per day, tire pressure should be checked prior to each flight as part of the preflight inspection.

A visual inspection of the tires is insufficient to determine whether or not they have acceptable tire pressure. When looking at a tire, it may appear to be properly inflated when it could, in fact, be significantly under-inflated. The only way to verify proper tire pressure is to measure the tire pressure with a quality, calibrated tire gauge.

Discounting tire growth after initial inflation, once the tire has been inflated, the maximum permissible pressure drop is 5% in any 24 hour period. When the loss rate exceeds 5% in 24 hours, check tires for damage.

Some aircraft tires are designed to permit air or nitrogen that is trapped in the cords or that diffuses through the liner or tube to escape through special sidewall vents. This venting prevents pressure build-up within the cord body which might cause tread, sidewall or ply separation. Tires requiring lower sidewall vents will have either a green or white paint dot applied to the area of each vent hole. Tires not needing lower sidewall vents will have no color dot in this zone.

When water or a soap solution is brushed over the outside of a vented tire, bubbles form. Some vents may emit a continuous stream of bubbles. Others may produce intermittent bubbles. And some may not bubble at all. This variety is normal and does not mean that there is anything wrong with the tire. In fact, as long as a vented tire is inflated, there will be some diffusion from the vents. Vents should remain open, so check periodically to make sure they have not been covered over or closed by tire paint or spilled solvent. And since vents may be covered during retreading, check for evidence that your retreads have been revented.

However, as stated above, any drop in pressure exceeding 5% in any 24 hour period is not acceptable.

(1) Several basic characteristics of aircraft tires may be mistaken for problems:

(a) Tire growth in the first 12 to 24 hours after inflation will result in a seemingly severe pressure drop. Simply inflate, wait for another 24 hours, then check pressure. It will probably be within specs.

(b) Make sure that initial inflation is to recommended operating pressure to ensure full tire growth.

(c) It is normal for tires to show a small amount of pressure leakage throughout the life of the tires.

(2) Maintain tires at pressure specified in 6-00-00, Chart 1. When checking tire pressure, examine tires for wear, cuts, bruises, and slippage.

NOTE: Use nitrogen, if available, to inflate tires. It will not sustain combustion and will reduce degradation, due to oxidation, of the liner material, casing plies, and wheel.

(3) Apply Age-Master #1 to tires to protect against ozone attack and weathering as follows:

(a) Clean oil and grease from all tire surfaces.

(b) Apply single heavy coat using brush at 0.4 - 0.5 fluid ounces per square foot. Cover surface completely and evenly; allow to dry for 5 - 10 minutes.

(c) Apply second coat per step (2); allow to dry for 20 - 30 minutes before handling.

(d) Remove agent on wheel assembly with cleaning solvent.

(e) Re-apply as conditions dictate.

4. Hydraulic System

The general condition of the hydraulic pump and landing gear actuating cylinders should be checked. Ensure that there are no leaks and that the line fittings are tight. The cylinder rods are to be free of all dirt and grit. To clean the rods use a rag soaked in hydraulic fluid and carefully wipe them. All the hydraulic lines should also be checked for leaks, kinks, and corrosion. Check the tightness of the attachment fittings.

Repair and inspection procedures for the hydraulic pump, cylinders and various components may be found in 29-10-00 of this manual.

Servicing Hydraulic Pump/Reservoir

(1) Parker Hannifin

The fluid level of reservoir of the combination pump and reservoir is checked every 50 hours by viewing fluid levels in the sight gauge. If fluid is below the fill-line, remove filler plug and fill with MIL-PRF-5606 hydraulic fluid (See “Chart 1” on page 12205). Install filler plug and tighten.

(2) Frisby/Triumph

Check fluid level in reservoir every 50 hours by viewing fluid level directly. If fluid is below the fill-line (i.e. - the “L” mark), remove fill plug and fill vent screw and add MIL-PRF-5606 hydraulic fluid (See “Chart 1” on page 12205) Ensure that the landing gear is down when filling the reservoir. Re-install fill plug and fill vent screw and tighten.

NOTE: Add clean filtered MIL-PRF-5606 (Revision “F” or higher) hydraulic fluid to the reservoir as necessary to keep fluid level just above the “L” mark on the sight gauge with the landing gear in the down and locked position. Verify that the fluid level does not exceed the “F” mark on the reservoir when the landing gear is in the up and locked position. Do not overfill. Add fluid only when level falls below the “L” mark. When filling, allow fluid level to stabilize before comparing fluid level to fill marks. Overfilling will result in excess fluid being expelled from reservoir vent pipe.

5. Engine Lubrication

CAUTION: DO NOT INTRODUCE ANY TRADE ADDITIVE TO THE BASIC LUBRICANT UNLESS RECOMMENDED BY THE ENGINE MANUFACTURER.

A. General

(1) The engine oil and oil filter should be changed every 50 hours of engine operation (or less, depending upon dust, humidity or engine conditions) or every 4 months, whichever comes first.

(2) For new, overhauled or re-manufactured engine, use ashless dispersant oil conforming to SAE-J-1899 specifications. After 25 hours, change oil and filter.

(3) It is recommended after the first 25 hours of engine operation that the operator continues to use ashless dispersant oil conforming to SAE-J-1899 specifications.

(4) The Lycoming engine is provided with a wet sump pressure oil system having a capacity of twelve (12) quarts. The minimum safe quantity of oil in the sump is 2-3/4 quarts. Single or multi viscosity aviation grade oils, in accordance with latest issue of Lycoming Service Instruction 1014, are required. (See 11-20-00, Figure 1, Placard #4)

NOTE: When operating temperatures overlap the indicated ranges, use the lighter grade oil.

(5) Refer to 81-20-00, Turbocharger Pre-Lubrication, for information on prelubricating the turbochargers after the engine oil change.

B. Draining Oil Sump

The engine should be warmed to operating temperature to ensure complete draining of the used oil.

An oil quick drain valve is located on the lower left side of the oil sump. To drain the oil, install a suitable length of inch I.D. hose on the outlet port. Remove the safety wire from the locking device, push “IN” and lock to begin draining. After all oil has drained from the sump, pull “OUT” and lock the outlet port closed. Safety wire the drain outlet port closed (“OUT” position). Remove the drain hose and service with the correct grade and amount of oil. Check for oil leakage and correct oil level after ground run-up.

C. Oil Filter - Full Flow

(1) Change the oil filter after the first 25 hours of engine operation for new, overhauled or re-manufactured engines and there after every 50 hours of engine operation; this is accomplished by removing the lockwire from the bolt head at the end of the filter, loosening the bolt, and removing the filter assembly from the adapter.

(2) Before discarding the filter element, remove the outer perforated paper cover and, using a sharp knife, cut through the folds of the element at both ends, close to the metal caps. Then, carefully unfold the pleated element and examine the material trapped in the filter for evidence of internal engine damage such as chips or particles from bearings. In new or newly overhauled engines, some particles of metallic shavings might be found, these are generally of no consequence and should not be confused with particles produced by impacting, abrasion or pressure. Evidence of internal engine damage found in the oil filter justifies further examination to determine the cause.

(3) Install new oil filter as follows:

(a) Before installing the new filter, lubricate the gasket on the filter with engine oil, then install it. Tighten slightly more than hand tight or to 1 full turn after gasket makes contact. Do not over torque.

(b) Run the engine and check for oil leaks; then install lockwire between nut on filter and oil filter adapter assembly.

D. Filling Oil Sump

The oil sump should normally be filled with oil to the 12 quart mark on the engine dipstick. The quantity of oil required for the engine may be found in 6-00-00. The specified grade of oil may be found on the oil filter access door or in the latest revision of Lycoming Service Instruction 1014. To service the engine with oil, open the access door and remove the oil filler cap.

6. Electrical System

Servicing the electrical system involves adding distilled water to the battery to maintain correct electrolyte level (Dry-Charged Lead-Acid - S/N's 4636001 thru 4636374), checking for any spilled electrolyte that would lead to corrosion and checking cable connections. The security of all electrical connections should be checked, as well as the operation of all lights, general condition of the alternator(s) and starter. All electrical wires should be inspected for chafing and bare wires. For detailed information on this system, refer to Chapter 24 of this manual.

A. Battery - Dry-Charged Lead-Acid

S/N's 4636001 thru 4636374

Access to the 24-volt battery is gained by opening the forward baggage door and removing the left floor of the forward baggage compartment.

The battery should be checked for proper fluid level. DO NOT fill the battery above the baffle plates. DO NOT fill the battery with acid - use water only. A hydrometer check will determine the percent of charge in the battery.

Inspect overflow sump for presence of battery fluid. Fluid in the sump is not a normal condition and indicates either a battery or charging system problem. If fluid is present, the electrical system must be serviced to eliminate the cause and the neutralizer media in the sump jar replaced.

If the battery is not up to charge, recharge starting at a 4-amp rate and finishing with a 2-amp rate. Quick charges are not recommended.

B. Battery - Valve-Regulated Lead-Acid

S/N's 4636375 and up and S/N's 4692001 and up

Access to the 24-volt battery is gained by opening the forward baggage door and removing the left floor of the forward baggage compartment.

Gill valve-regulated non-maintainable batteries are fully charged when they leave the factory. Non-maintainable means that this type of battery cannot be opened to add fluid. Periodic maintenance is still required in the form of boost charging if the battery becomes discharged.

7. Power Plant

Regularly check the engine compartment for oil and fuel leaks, chafing of lines, loose wires and tightness of all parts. Maintenance instructions for the power plant may be found in Chapter 71 of this manual and in the appropriate manufacturer's manuals.

8. Propeller

The propellers should be inspected for grease or oil leakage and freedom of rotation on the hub pilot tube. To check freedom of rotation, rock the blade back and forth through the slight freedom allowed by the pitch change mechanism. Lubricate the propeller at 100-hour intervals per the appropriate Hartzell Owner's Manual.

Additional service information for the propeller may be found in 61-10-00.

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SCHEDULED SERVICING

Routine cleaning and lubrication of the airplane and its component parts will significantly extend its service life and reduce the frequency of repairs.

1. Induction Air Filter

The induction air filter is located in the forward portion of the lower engine cowls.

A. Removal

(1) Remove the induction air panel from the nose cowl and from the lower engine cowls.

(2) Remove the filter assembly from the induction box assembly.

(3) Clean or replace the filter.

B. Installation

After cleaning and inspection, install the filter element and cover in reverse order of removal instructions.

C. Cleaning and Inspection

(1) The filter should be cleaned daily when operating in dusty conditions, and if any holes or tears are noticed, the filter should be replaced immediately. For replacement filter, refer to Parts Catalog.

(2) Remove the filter element and blow out with compressed air from gasket side or wash in warm water and mild detergent and allow to dry.

(3) The filter housing can be cleaned by wiping with a clean cloth soaked in suitable quick drying type solvent. When the housing is dry, reinstall in accordance with Installation.

2. Alternate Air Door

The alternate door is located in the air induction box to provide a source of air to the engine should there be an air stoppage through the filter system. The following should be checked during inspection:

A. Check that air door seals are tight and that the hinge is secure.

B. Check that when the cockpit control is in the closed position the door is properly seated in the closed position.

C. Actuate the door by operating the control lever in the cockpit to determine that it is not sticking or binding.

D. Check the cockpit control cable for free travel.

3. Propeller

Inspect the spinner, back plate and propeller surfaces for nicks, scratches, corrosion and cracks. Remove minor nicks and scratches per instructions in 61-10-00 or the appropriate Hartzell Blade Manual (for composite blades see latest revision Hartzell Composite Blade Manual No. 135). The face of each blade should be painted when necessary with a flat paint to retard glare. To prevent corrosion, wipe surfaces with a light oil or wax.

4. Tire Balance

Proper balancing is critical for the life of aircraft tires. If a new tire is balanced upon installation it will usually remain balanced for the life of the tire without having any shimmy or flat spots.

A. TIre Balancer

An inexpensive balancing fixture that will balance almost any light aircraft tire can be made from the materials shown in Figure 1.

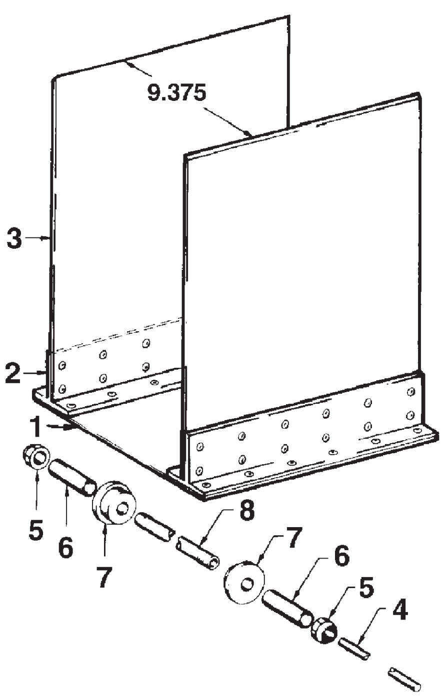
(1) Chamfer top edges of -3 sides, leaving 1/16 inch flat on top of the inboard edge. Rivet -2 tees to -3 sides using AN 470-AD5 rivets, with 2 inch spacing, and using AN 426-AD5 rivets (2 inch center to center) to secure -2 tee's to -1 base. If tee extrusion is unavailable, heavy angle extrusion could be used. -3 sides must be vertical.

(2) The -4 axle must slide through the -8 pipe, the -5 nuts are made by reaming the existing threads in the AN 365-624 nuts with an R drill, then tapping them with a 1/8-27 pipe tap.

(3) The -6 spacers were made from 1/2 inch aluminum tubing, the two lengths of spacers are suitable for balancing most any aircraft wheel.

(4) The -7 bushings may be made from one inch phenolic or aluminum using a 1-1/2 inch hole saw to cut out the smaller bushing and a 1-3/4 hole saw to cut out the larger. By inserting a 1/4 inch long threaded bolt through the pilot hole and securing with a washer and nut, a drill press and file may be used to make the off-set on the bushing. The turned-down part should just slide inside the bearing race and then ream the pilot hole to slide over the -8 pipe threads.

(5) The -8 pipe was made from a piece of 1/8 inch black pipe and threaded with a 1/8-27 pipe die, this will be thread 3 inches in from each end of the pipe.



USE THE FOLLOWING LIST OF MATERIALS TO MAKE THE BALANCER

- 1. 1 EA BASE	12 X 11	0.190 2024 T3 CLAD ALUMINUM ALLOY
- 2. 2 EA TEE	2.5 X 2 X 11	0.190 2024 T4 EXTRUDED ALUMINUM ALLOY
- 3. 2 EA SIDES	14 X 11	0128 2024 T3 CLAD ALUMINUM ALLOY
- 4. 2 EA AXLE	0.125 X 10.25	4130 STEEL, NOMALIZED
- 5. 2 EA NUTS	AN 365-624	
- 6. 2 EA SPACER	0.50 X 2.25	5052-0 ALUMINUM TUBING
2 EA SPACER	0.50 X 1.25	5052-0 ALUMINUM TUBING
- 7. 2 EA BUSHING	1.480 X 1.625 X 1.00	PHENOLIC OR ALUMINUM
2 EA BUSHING	2.240 X 1.37 X 1.00	PHENOLIC OR ALUMINUM
- 8. 1 EA PIPE	1/8 X 9.3	BLACK STEEL PIPE
* 2 EA BEARINGS	SAVE TWO OF EACH SIZE WORN WHEEL BEARINGS FROM PREVIOUS INSPECTIONS.	

Tire Balancer
Figure 1

B. Procedure

Balance tires as follows:

(1) Mount the tire and tube (if one is used) on the wheels, but do not install the securing bolts. Install the wheel bearings in the wheels; then, using the -7 bushings, -6 spacers, and -5 nuts, install the wheel/tire assembly on the -8 pipe. Secure the -5 nuts finger tight so that the wheel halves touch each other. Be sure the bolt holes are aligned. Insert the -4 axle through the -8 pipe and place the wheel in the center of the balancer. Make sure the axle is only on the chamfered edges of the balancer and that it is at 90° to the sides of the balancer.

(2) Release the tire. If it is out of balance it will rotate, coming to rest with the heaviest point on the bottom. Tape a ounce patch across the top center of the tire. Rotate the tire 45° and release it again. If the tire returns to the same position, add a 1 ounce patch and again rotate the tire and release it. Continue this procedure until the tire is balanced.

(3) When balance is attained, put a chalk mark on the sidewall directly below the patch. Use one mark for each half ounce of weight needed. Mark the valve stem location on the tire and the opposite wheel half to assure reassembly in the same position. Remove the wheel from the balance stand, break it down and clean the inside of the tire with toluol. Apply a coat of patch cement to both the patch and the inside center of the tire in line with the chalk marks. When the cement has dried, install the patches making certain they are on the centerline of the tire and aligned with the chalk marks on the sidewall. Burnish the patches to remove trapped air, etc.

(4) When reassembling the wheel, powder the inside of the tire. Mount the tire on the valve side of the wheel in the same position it was in when it was balanced. Install the other wheel half, aligning the chalk marks. Install the bolts and tighten to required torque, then inflate the tire to the pressure specified in Chart 1, 6-00-00. and recheck the balance. The wheel should not be more than ounce out of balance.

5. Lubrication

Proper lubrication procedures are valuable both as a means of prolonging the service life of the airplane and reducing the frequency of repairs. The periodic application of recommended lubricants to their relevant bearing surfaces, together with cleanliness, ensures the maximum efficiency and utmost service life of all moving parts. Instructions regarding the locations, time intervals, and types of lubricant used are found in the Lubrication Charts. See also 91-10-00, Chart 10, Consumable Materials.

NOTE: If the airplane is inactive for long periods of time, lubricate in accordance with Lubrication Charts at least every 90 days.

A. Precautions

To ensure the best possible results from the application of lubricants, observe the following precautions:

CAUTION: MIL-PRF-23827 AND MIL-PRF-81322, CONTAIN CHEMICALS WHICH MAY BE HARMFUL TO PAINTED SURFACES.

CAUTION: DRY LUBRICANT (I.E. - PTFE BASED MS-122) WILL ATTACK ANY ACRYLIC BASED PLASTIC (LUCITE), POLYCARBONATES (LEXAN), POLYSTYRENE AND ITS COPOLYMERS (ABS), AND CELLULOSE ACETATE.

CAUTION: AFTER THOROUGHLY WASHING AIRPLANE, ENSURE LANDING GEAR, FLIGHT CONTROLS, FLAP TRACKS, STABILATOR TRIM SCREW, AND ENGINE COMPARTMENT ARE STILL PROPERLY LUBRICATED.

(1) Use recommended lubricants. Where general purpose lubricating oil is specified, but unavailable, clean engine oil may be used.

(2) Check components to be lubricated for evidence of excessive wear and replace as required.

(3) Remove excess lubricant from components to prevent collecting dirt and sand in quantities capable of causing excess wear or damage to bearing surfaces.

B. Application of Grease

Before using a grease gun, ensure that gun is filled with new, clean grease of the grade specified for the particular application.

(1) Where a reservoir is not provided around a bearing, apply the lubricant sparingly and wipe off any excess.

(2) Remove wheel bearings from the wheel hub and clean thoroughly with a suitable solvent. When repacking with grease, be sure the lubricant enters the space between the rollers in the retainer ring. Do not pack the grease into the wheel hub.

(3) Use extra care when greasing the constant speed propeller hub to avoid blowing the clamp gaskets. Remove one grease fitting and apply grease to the other fitting until fresh grease appears at the hole of the removed fitting.

C. Application of Oil

Whenever specific instructions for lubrication of mechanisms requiring lubrication are not available, observe the following precautions:

(1) Apply oil sparingly, never more than enough to coat the bearing surfaces.

(2) Do not oil control cables.

D. Lubrication Charts

(PIR-89506-002 Rev B.)
(PIR-PPS65102-134, Rev. New.)

The lubrication charts consist of individual illustrations for the various aircraft systems. Each component to be lubricated is indicated by a number, the type of lubricant and the frequency of application. Special instructions are listed in “Chart 2” before the lubrication charts.

NOTE: When the average ambient air temperature is approximately at the dividing line, use the lighter oil.

While the specified lubricant should be used if available, lubricants listed in the Lubrication Charts should all be considered “or equivalent.” Lubricant specifications become problematic over time. Where a specific product is called out, that manufacturer may go out of business, may be sold, or renamed. The named product may subsequently be no longer available, or renamed itself. Many lubricant military specifications have been superceded over the last several years. Accordingly, a cross-reference chart (“Chart 1”) is provided for lubricants where specification or product changes have been identified.

CHART 1
LUBRICANT SPECIFICATION CROSS-REFERENCE (Sheet 1 of 2)

Old Spec / Product superceded by	**New Spec / Product**	**Product Type**
Lubriplate 907	Lubriplate Aero	Grease, Lithium Base, Multipurpose
MIL-A-907	MIL-PRF-907 (aka Kopr-Kote *)	Anti-Seize Thread Compound, High Temp. (up to 566 Degrees C) (up to 1050 Degrees F)
MIL-A-5092	MMM-A-1617	Adhesive, (Rubber Base) General Purpose.
MIL-C-16173	MIL-PRF-16173	Corrosion Preventative Compound, Solvent Cutback, Cold Application.
MIL-DTL-27686	See MIL-I-27686 below.	
MIL-G-3278	MIL-PRF-23827	Grease, Aircraft & Instrument, Gear and Actuator Screw.
MIL-G-3545	MIL-PRF-81322	Grease, Aircraft, General Purpose, Wide Temp. (-54 to 177 Degrees C) (-65 to 350 Degrees F).
MIL-G-6032	SAE-AMS-G-6032	Grease, Plug Valve, Gasoline & Oil Resistant.
MIL-G-7711	MIL-PRF-81322	See MIL-PRF-81322 above.
MIL-G-18709	DOD-G-24508	Grease, High Perfomance, Multipurpose.
MIL-G-21164	MIL-G-21164	Grease, Molybdenum Disulfide, for Low & High Temperatures
MIL-G-23827	MIL-PRF-23827	See MIL-PRF-23827 above.
MIL-G-81322	MIL-PRF-81322	See MIL-PRF-81322 above.
MIL-H-5606	MIL-PRF-5606	Hydraulic Fluid, Petroleum Base, Aircraft, Missile, and Ordnance.
MIL-I-27686	MIL-DTL-85470	Inhibitor, Icing, Fuel System, High Flash, NATO Cold Number S-1745.
MIL-L-6082	SAE J 1966	Oil, Lubricating, Aircraft Piston Engine (Non-Dispersant Mineral Oil).
MIL-L-7870	MIL-PRF-7870	Oil, Lubricating, Low Temp.
MIL-L-22851	SAE J 1899	Oil, Lubricating, Aircraft Piston Engine (Ashless Dispersant).
MIL-L-25567	MIL-PRF-25567	Leak Detection Compound, Oxygen Systems.
MIL-L-60326	MS-122AD *	Dry-Lubricant.
MIL-M-7866	SAE-AMS-M-7866	Molybdenum Disulfide, Technical, Lubrication Grade.

CHART 1
LUBRICANT SPECIFICATION CROSS-REFERENCE (Sheet 2 of 2)

Old Spec / Product	superceded by	**New Spec / Product**	**Product Type**
MIL-S-11031B		A-A-59293	Adhesive (Curing), Sealing Compound (Polysulfide Base).
MIL-S-22473		ASTM-D-5363	Adhesive, Anaerobic Single-Component.
MIL-S-8660		SAE-AS-8660	Silicone Compound, Nato S-736, (-54 to 204 Degrees C) (-65 to 400 Degrees F)
MIL-T-5544		SAE-AMS-2518	Thread Compound, Anti-Seize, Graphite-Petrolatum
MIL-T-27730		A-A-58092	Tape, Anti-Seize, Polytetrafluoroethylene
MS-122 *		MS-122AD *	Dry-Lubricant
MS-122-6075 *		MS-122AD *	Dry-Lubricant
Parker O-Ring Lube *		Parker O-LUBE *	O-Ring Lubricant
Parker 6PB * or 6PB Parker *		MIL-PRF-907 (aka Kopr-Kote *)	See MIL-PRF-907 above.
TT-A-580 (JAN-A-669)		TT-A-580 (TT-S-1732)	Sealing Compound, Pipe Joint and Thread, Lead Free, General Purpose.
* Product Nomenclature			

CHART 2
SPECIAL INSTRUCTIONS

1. BEARINGS AND BUSHINGS - Clean exterior with a quick-drying solvent before lubricating.
2. LUBRICATION POINTS - Wipe all lubrication points clean of old grease, oil, dirt, etc., before lubricating.
3. WHEEL BEARINGS - For nose and main wheels, see bearing installation procedure in 32-40-00, under Main Wheel Assembly, Assembly and Installation, for grease requirements and packing instructions. Wheel bearings also require cleaning and repacking after exposure to any abnormal quantity of water.
4. OLEO STRUTS, HYDRAULIC PUMP RESERVOIR AND BRAKE RESERVOIR - Fill per instructions on unit or container, or refer to applicable chapter in this manual.
5. DOOR SEALS - Apply release agent/dry lubricant to door seals at least once a month to improve sealing characteristics and to prevent the seal from sticking.
6. CONTROL CABLES - Do not oil control cables. Grease control cables where they pass over a pulley or through a fairlead.
7. AIR FILTER - To clean filter, tap gently to remove dirt particles or wash in warm water and mild detergent and dry. Do not blow out with compressed air. Do not use oil. Replace filter if damaged.
8. OIL AND FILTER - Lycoming recommends changing the oil and filter every 50 hours or four months, whichever comes first.
9. POWER PLANT - See latest revision of Lycoming Service Instruction No. 1014 for use of detergent oil.
10. CONTROL WHEEL SHAFT BUSHING - Do not lubricate. Clean only, using alcohol or other suitable solvent.
CAUTION: GREASE MUST BE APPLIED TO ALL BLADES OF A PROPELLER ASSEMBLY AT THE TIME OF LUBRICATION. CAUTION: DO NOT MIX GREASES. MIXING NYCO GN 3058 WITH OTHER GREASES IS NOT PERMITTED. IF NYCO GN 3058 IS INADVERTENTLY MIXED WITH ANOTHER GREASE, REMOVE THE PROPELLER FROM THE AIRCRAFT: DISASSEMBLE, CLEAN, AND SERVICE WITH ALL NEW GREASE. 11. PROPELLER - See the appropriate Hartzell Propeller Owner’s Manual for lubrication procedure. If annual usage is significantly less than 100 hours, increase lubrication frequency to every six months. See also latest revision of Hartzell Manual No. 202A. If initial lubrication after assembly or overhaul, use procedure in Hartzell Manual No. 202A.
12. NOT USED.
13. FLAP ACTUATOR SCREWJACK - If equipped with grease fittings, use a grease gun and lubricate with flaps up. If no grease fittings installed, extend flaps to full down and brush coat grease on exposed screwjack threads.
14. DOWNSPRING AND ROD ASSEMBLY - Lubricate annually between Downspring and Road Assy., and lubricate sliding components of Rod Assy. See disassembly procedure in 32-30-00.
15. BUNGEE - Lubricate springs if bungee is disassembled.

CHART 3
LUBRICATION CHART (Cabin Door, Baggage Door, and Seats)

	COMPONENT	LUBRICANT	FREQUENCY
1.	BAGGAGE DOOR HINGE (Special Instruction #2)	MIL-PRF-7870C	100 HRS
2.	BAGGAGE DOOR LATCH MECHANISM (Special Instruction #2)	DRY LUBRICANT	100 HRS
3.	PILOT AND COPILOT SEAT ADJUSTMENTS	MIL-PRF-7870C	100 HRS
4.	CABIN DOOR HINGES (Special Instruction #2)	MIL-PRF-7870C	100 HRS
5.	CABIN DOOR LATCH MECHANISM (Special Instruction #2)	DRY LUBRICANT	100 HRS
6.	ACCESS DOOR - HINGE * (Special Instruction #2)	MIL-PRF-7870C	100 HRS
7.	ACCESS DOOR - LATCH * (Special Instruction #2)	DRY LUBRICANT	100 HRS

* If installed.

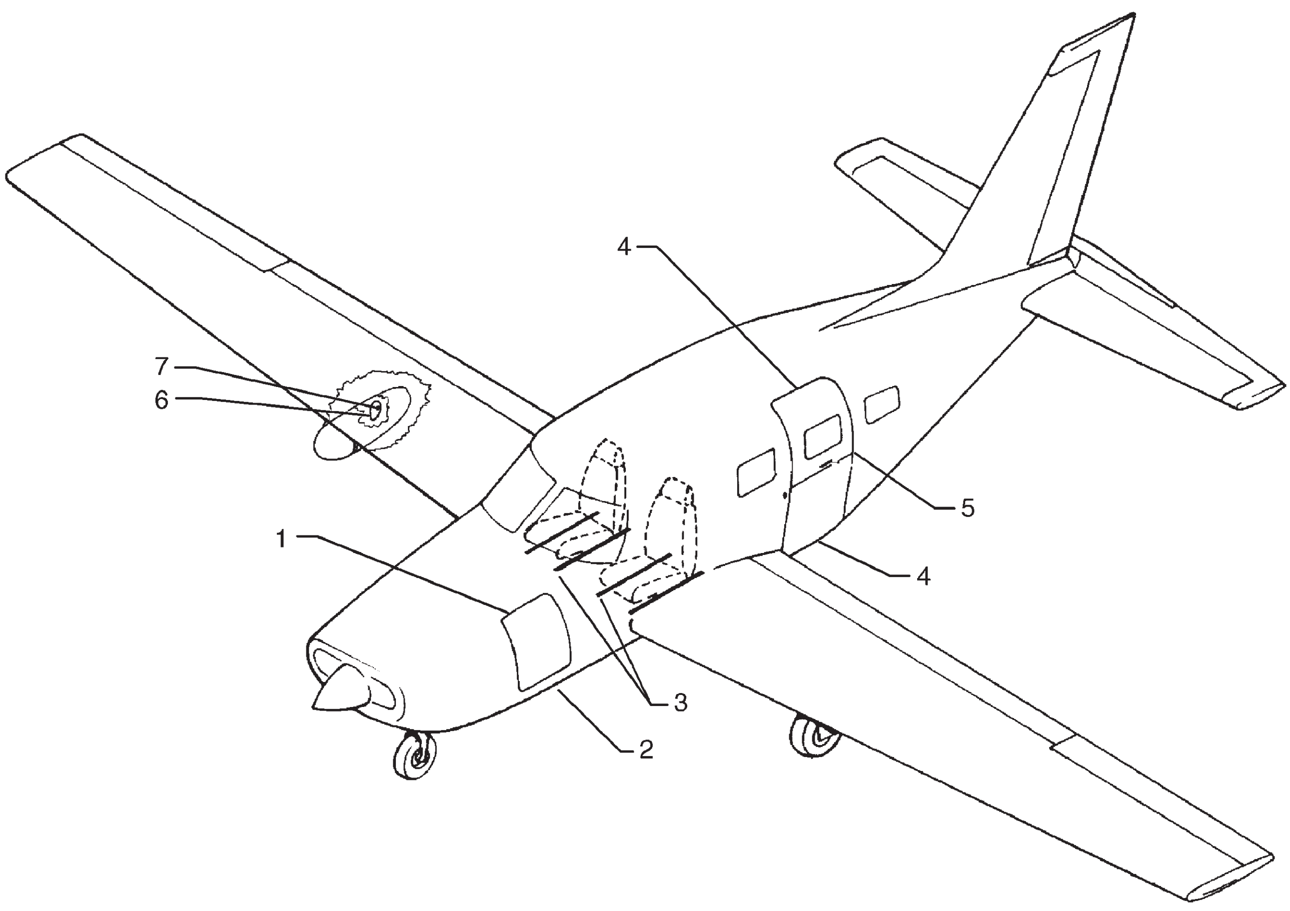


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CHART 4
LUBRICATION CHART (Main Landing Gear and Hydraulic System)

	COMPONENT	LUBRICANT	FREQUENCY
1.	BRAKE RESERVOIR (Special Instruction #4)	MIL-PRF-5606	50 HRS
2.	HYDRAULIC PUMP/RESERVOIR (Special Instruction #4)	MIL-PRF-5606	50 HRS
3.	RETRACTION CYLINDER ROD END BEARINGS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
4.	MAIN GEAR TORQUE LINKS (Special Instruction #2)	MIL-PRF-7870C	100 HRS
5.	MAIN GEAR WHEEL BEARINGS (Special Instruction #3)	MOBIL AVIATION GREASE SHC 100	100 HRS

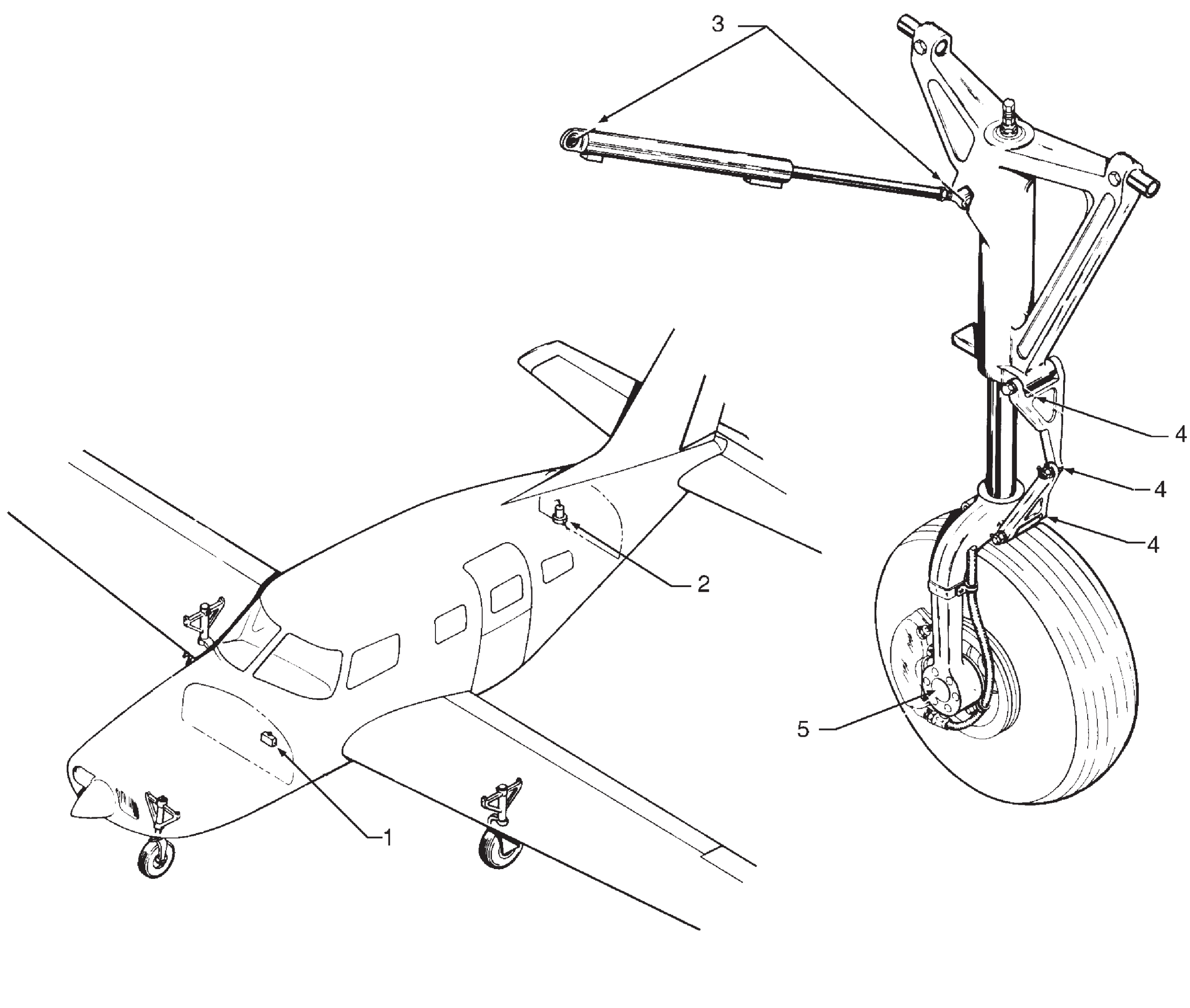


CHART 5
LUBRICATION CHART (Nose Gear)

COMPONENT	LUBRICANT	FREQUENCY
1. STEERING BELLCRANK PIVOT POINTS AND ROD ENDS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
2. RETRACTION CYLINDER AND ASSIST SPRING ROD END BEARINGS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
3. NOSE GEAR PIVOT POINT (Special Instruction #2)	MIL-PRF-81322	100 HRS
4. TORQUE LINK ASSEMBLY (Special Instruction #2)	MIL-PRF-7870C	100 HRS
5. NOSE WHEEL BEARINGS (Special Instruction #3)	MOBIL AVIATION GREASE SHC 100	100 HRS
6. STEERING BELLCRANK AND ROD ENDS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
7. NOSE GEAR DOOR HINGE (Special Instruction #2)	MIL-PRF-7870C	100 HRS
8. BUNGEE (Special Instructions #2 and 15)	Lubriplate Aero	As Required
9. DOWNSPRING AND ROD ASSEMBLY (Special Instruction #14)	MIL-PRF-23827	12 MONTHS

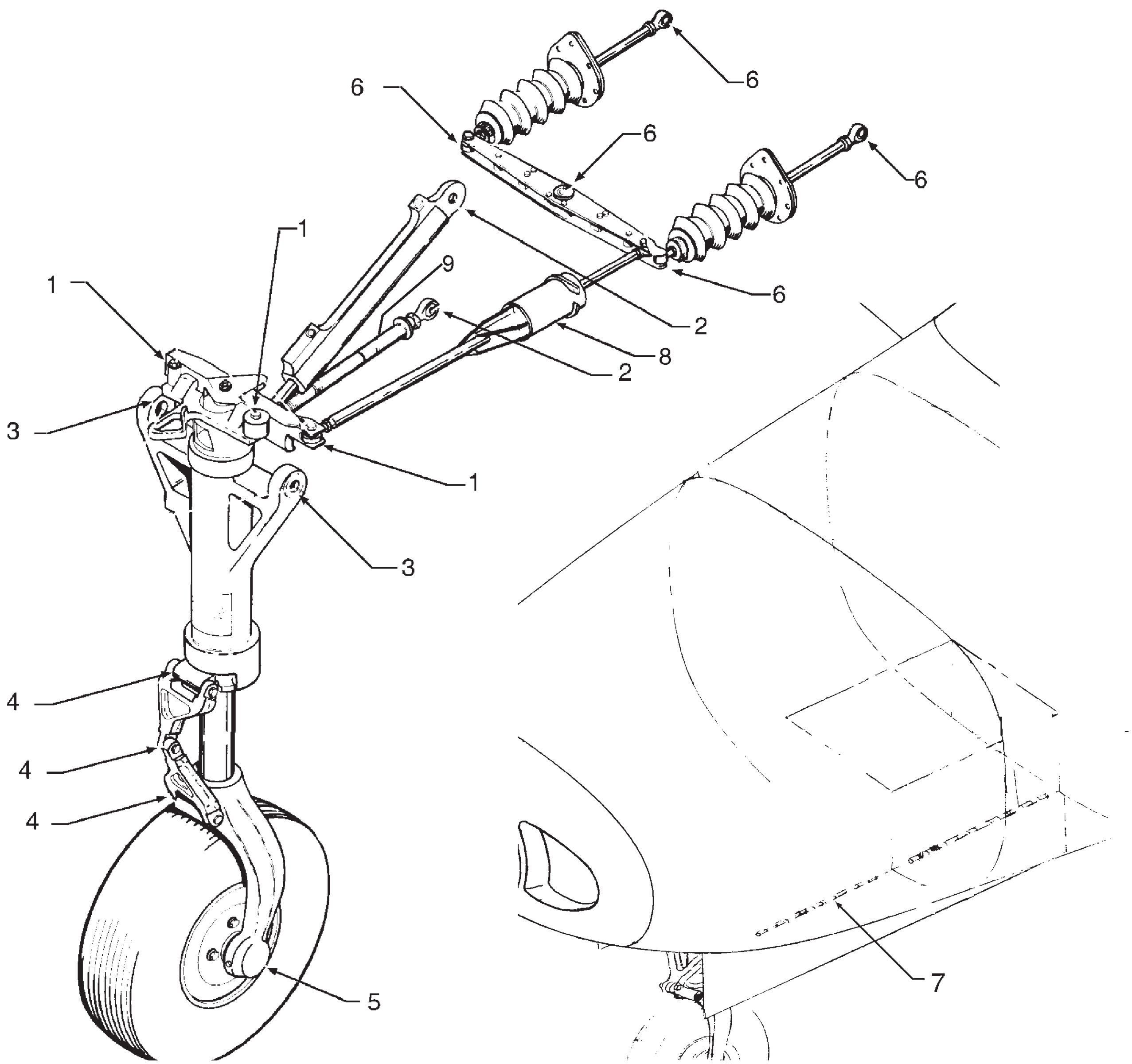


CHART 6
LUBRICATION CHART (Control System - Sheet 1 of 3)

	COMPONENT	LUBRICANT	FREQUENCY
	FLAP TRACKS AND ROLLERS (Special Instruction #2) (Not shown)	MIL-PRF-23827 or Lubriplate Aero	100 HRS
1.	FLAP CONTROL ROD END BEARINGS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
1A.	FLAP ACTUATOR SCREW JACK (Special Instruction #13)	Lubriplate Aero	100 HRS
2.	RUDDER CONTROL SECTOR (Special Instruction #1)	MIL-PRF-7870C	100 HRS
3.	ELEVATOR TRIM ROD END BEARINGS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
4.	ELEVATOR SECTOR AND CONTROL ROD END BEARINGS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
5.	ELEVATOR TRIM SCREW (Special Instruction #2)	MIL-PRF-23827	100 HRS

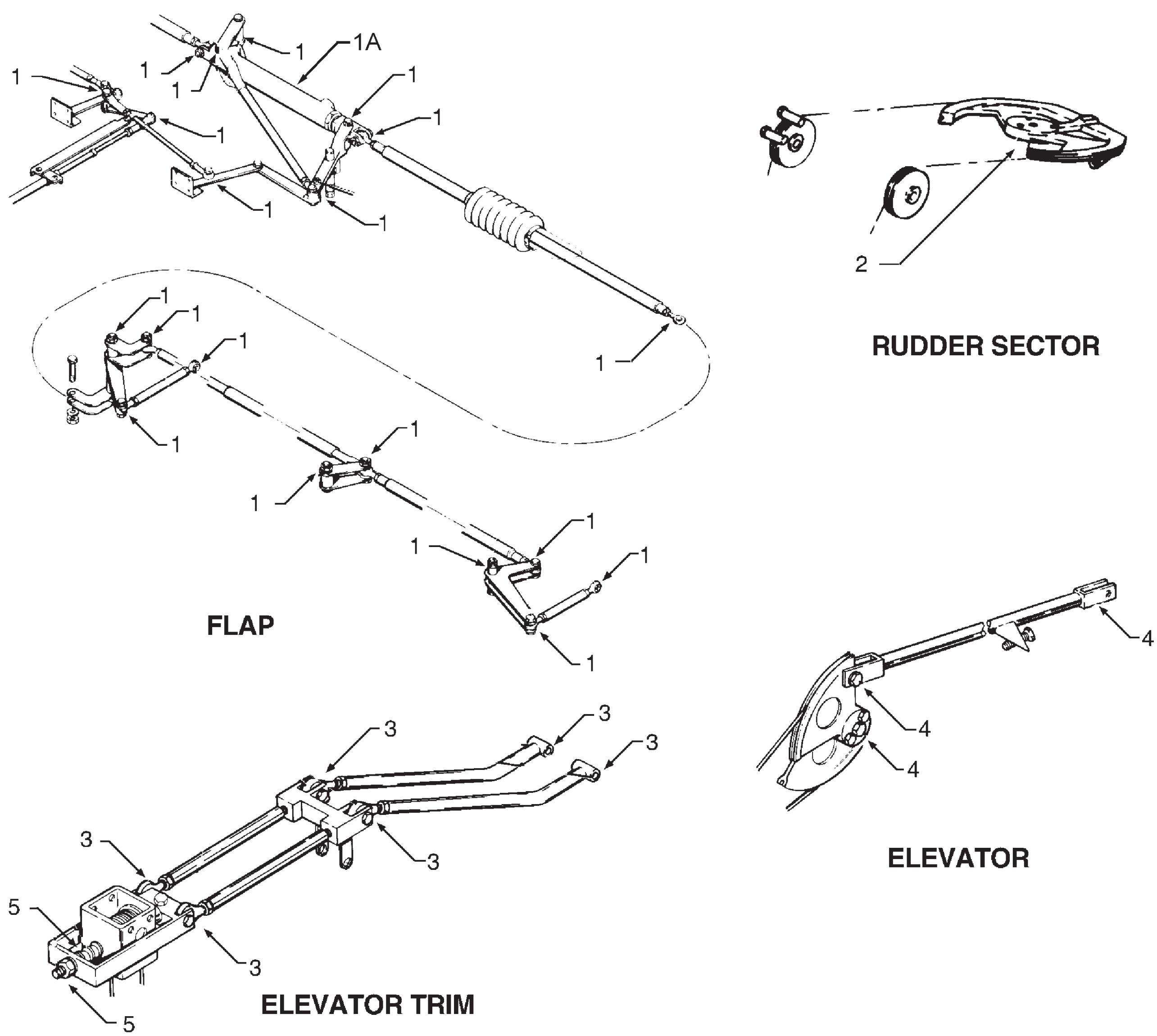
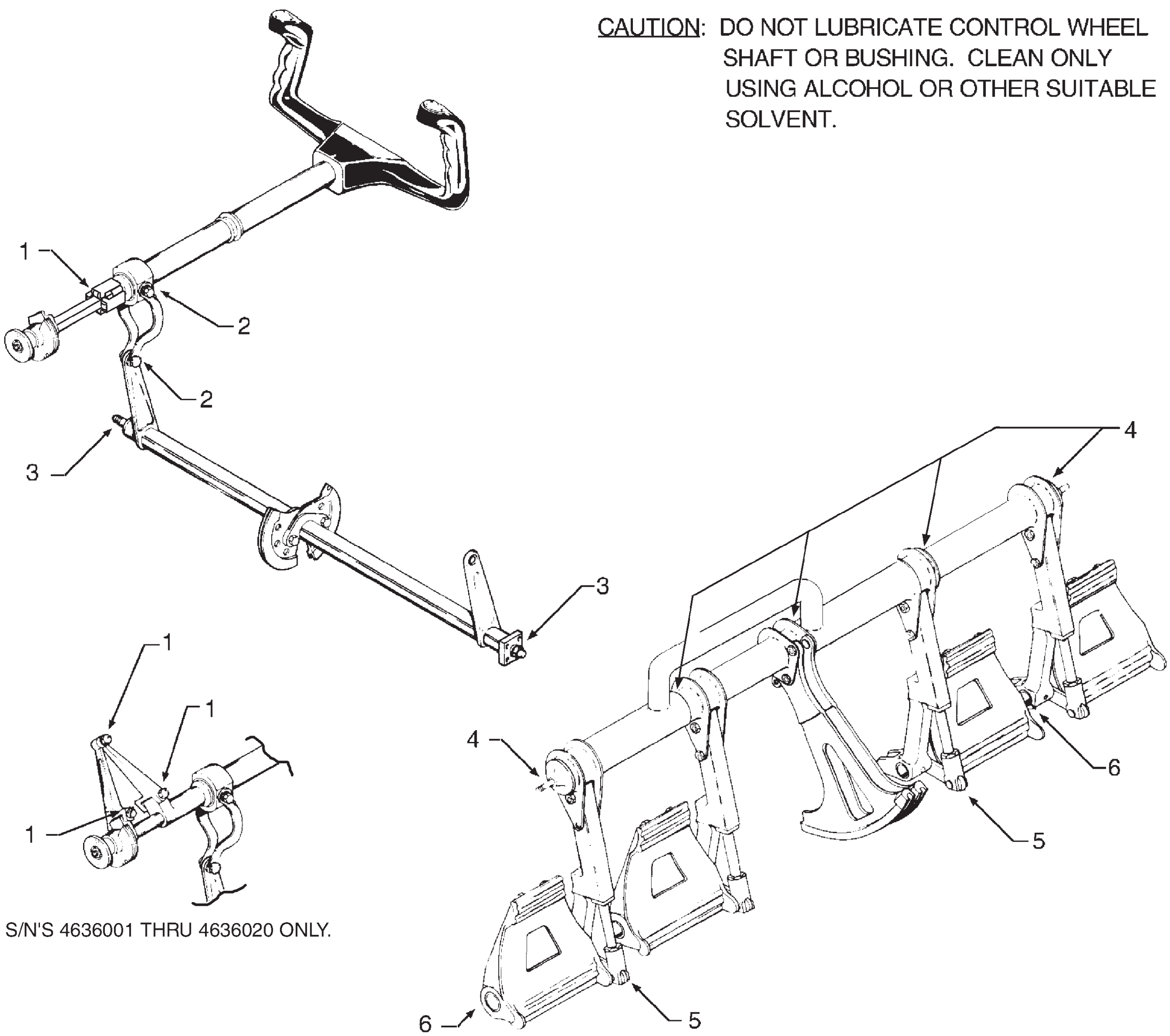


CHART 6
LUBRICATION CHART (Control System - Sheet 2 of 3)

	COMPONENT	LUBRICANT	FREQUENCY
1.	AILERON TORQUE ROLLER ASSEMBLY (Special Instruction #1)	MIL-PRF-7870C	100 HRS
1.	AILERON TORQUE ARM ASSEMBLY (Special Instruction #1)	MIL-PRF-7870C	100 HRS
2.	CONTROL COLUMN ATTACHMENT ARM (Special Instruction #1)	MIL-PRF-7870C	100 HRS
3.	ELEVATOR TORQUE TUBE ASSEMBLY (Special Instruction #1)	MIL-PRF-7870C	100 HRS
4.	RUDDER TORQUE TUBE ASSEMBLY (Special Instruction #2)	MIL-PRF-7870C	100 HRS
5.	BRAKE ROD ENDS (Special Instruction #2)	MIL-PRF-7870C	100 HRS
6.	BUSHING, RUDDER PEDAL MOUNT (Special Instruction #1)	MIL-PRF-7870C	100 HRS
*	AILERON, ELEVATOR, RUDDER AND TRIM CABLES (Special Instruction #2) (Grease cables where they pass through a fairlead or over a pulley.) (Not shown.)	MIL-PRF-23827	100 HRS

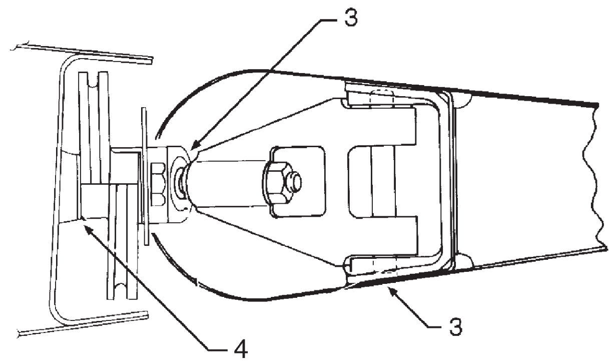
CAUTION: DO NOT LUBRICATE CONTROL WHEEL SHAFT OR BUSHING. CLEAN ONLY USING ALCOHOL OR OTHER SUITABLE SOLVENT.

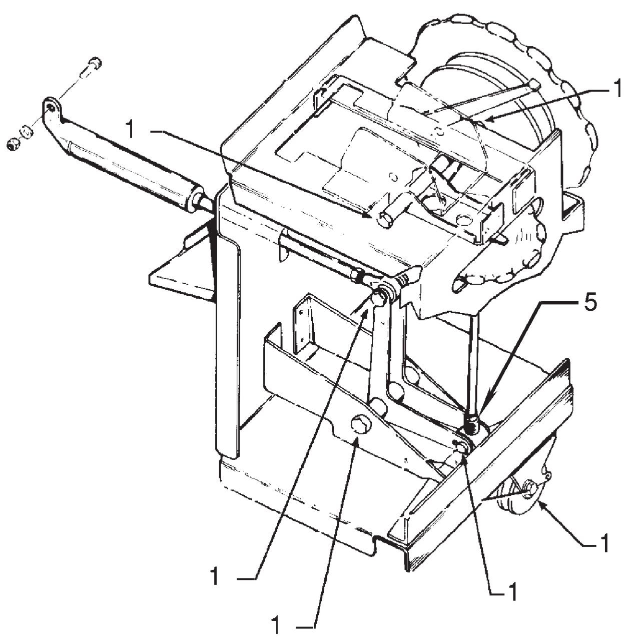


S/N'S 4636001 THRU 4636020 ONLY.

CHART 6
LUBRICATION CHART (Control System - Sheet 3 of 3)

	COMPONENT	LUBRICANT	FREQUENCY
1.	PULLEYS AND TRIM WHEELS (Special Instruction #1)	MIL-PRF-7870C	100 HRS
2.	AILERON HINGES (Special Instruction #1)	MIL-PRF-7870C	100 HRS
3.	AILERON SECTOR (Special Instruction #2)	DRY LUBRICANT such as MS-122AD	100 HRS
4.	AILERON SECTOR BEARING (Special Instruction #1)	MIL-PRF-7870	100 HRS /Annually
5.	RUDDER TRIM SCREW AND FOLLOWER (Special Instruction #2)	Lubriplate Aero	100 HRS
*	AILERON, ELEVATOR, RUDDER AND TRIM CABLES (Special Instruction #2) (Grease cables where they pass through a fairlead or over a pulley.) (Not shown.)	MIL-PRF-23827	100 HRS





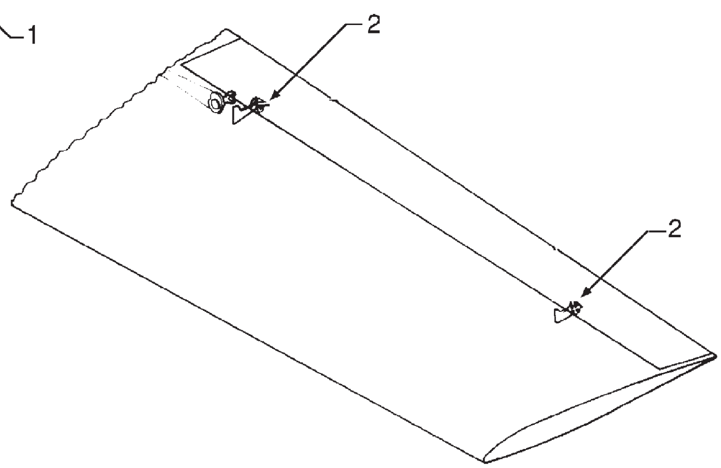
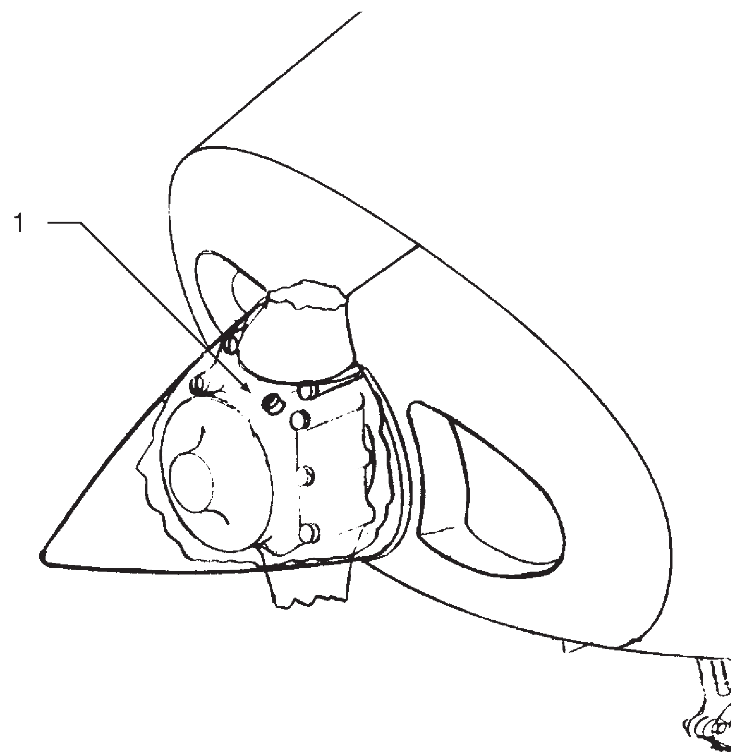


CHART 7
LUBRICATION CHART (Propeller)

COMPONENT	LUBRICANT	FREQUENCY
1. COMPOSITE BLADE PROPELLERS (Special Instruction #11)	See Note	100 HRS / 12 MONTHS (whichever comes first)
1. ALUMINUM BLADE PROPELLERS (Special Instruction #11)	See Note	100 HRS / 12 MONTHS (whichever comes first)
NOTE: Use grease specified on label on propeller.		



CHAPTER

20

STANDARD PRACTICES - AIRFRAME

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CHAPTER 20

LIST OF EFFECTIVE PAGES

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CHAPTER 20 - STANDARD PRACTICES - AIRFRAME

TABLE OF CONTENTS

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GENERAL

1. Description

This chapter contains general information pertaining to standard aircraft hardware installation and removal practices, as well as general information on corrosion control and painting issues.

For standard repair practices of a minor nature, see 51-70-00 and AC 43.13-1 (latest revision).

If non-destructive testing is needed after repair of 4130 steel, use a magnetic particle inspection method such as Magnaflux.

Testing and inspecting of aluminum castings and machined aluminum parts may be done by the dye penetrant method.

Usually, a good visual inspection with a 10X magnifying glass will show any damage or defect in a repair that is of a significant nature.

2. Torque Wrenches

Torque wrenches should be checked daily and calibrated by means of weights and a measured lever arm to ensure that inaccuracies are not present. Checking one torque wrench against another is not sufficient and is not recommended. Some wrenches are quite sensitive to the way they are supported during a tightening operation. Any instructions furnished by the manufacturer must be followed explicitly.

When it is necessary to use a special extension or adapter wrench together with a torque wrench, a simple mathematical equation must be worked out to arrive at the correct torque reading. Following is the formula to be used: (Refer to”Figure 1” on page 20002.)

T = Torque desired at the part.

A = Basic lever length from center of wrench shank to center of handle or stamped on wrench or listed for that model wrench.

B = Length of adapter extension, center of bolt to center of shank.

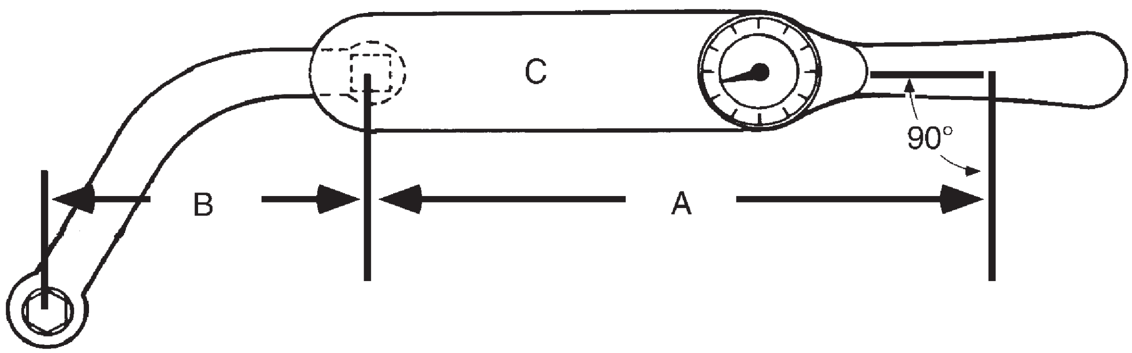
C = Scale reading needed to obtain desired torque (T).

The formula: $$C = \frac{A \times T}{A + B}$$

EXAMPLE: A bolt requires 30 foot pounds and a 3 inch adapter (one-quarter of a foot or 0.25’) is needed to get at it. You want to know what scale reading it will take on a one-foot lever arm wrench to obtain the 30 foot pounds at the bolt.

$$C = \frac{1 \times 30}{1 + 0.25} \quad \text{or} \quad C = \frac{30}{1.25}$$

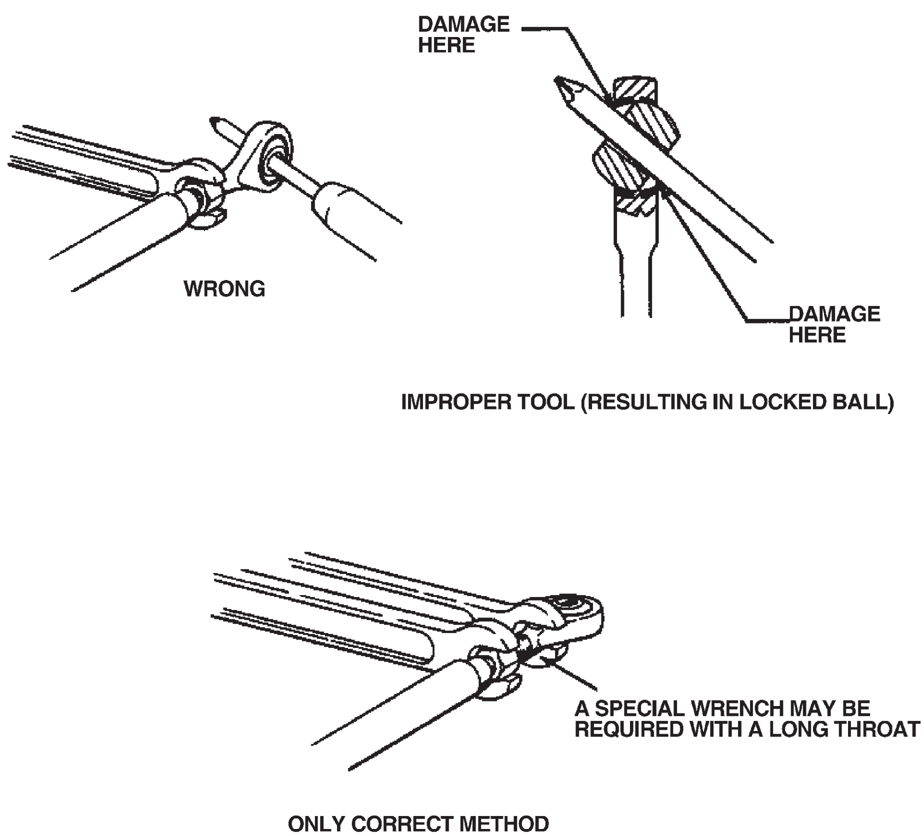
Remember, the 3 inch adapter must be projecting 3 inches straight along the wrench axis. In general, avoid all complex assemblages or adapters and extensions of flex joints.



Torque Wrench Formula
Figure 1

3. Installing Rod End Bearings

Install rod end bearings as shown in Figure 2.



Installing Rod End Bearings
Figure 2

4. Removing Cherrylock Rivets

Use following procedure to remove cherrylock rivets:

A. To remove from thick material, use a tapered steel drift pin to drive out rivet stem. (See "Figure 3", View 1.)

CAUTION: DRIVING OUT THE LOCKED STEM OF RIVETS INSTALLED IN THIN MATERIAL MAY DAMAGE THE MATERIAL.

NOTE: Drilling completely through the rivet sleeve, when removing rivets, tends to enlarge hole.

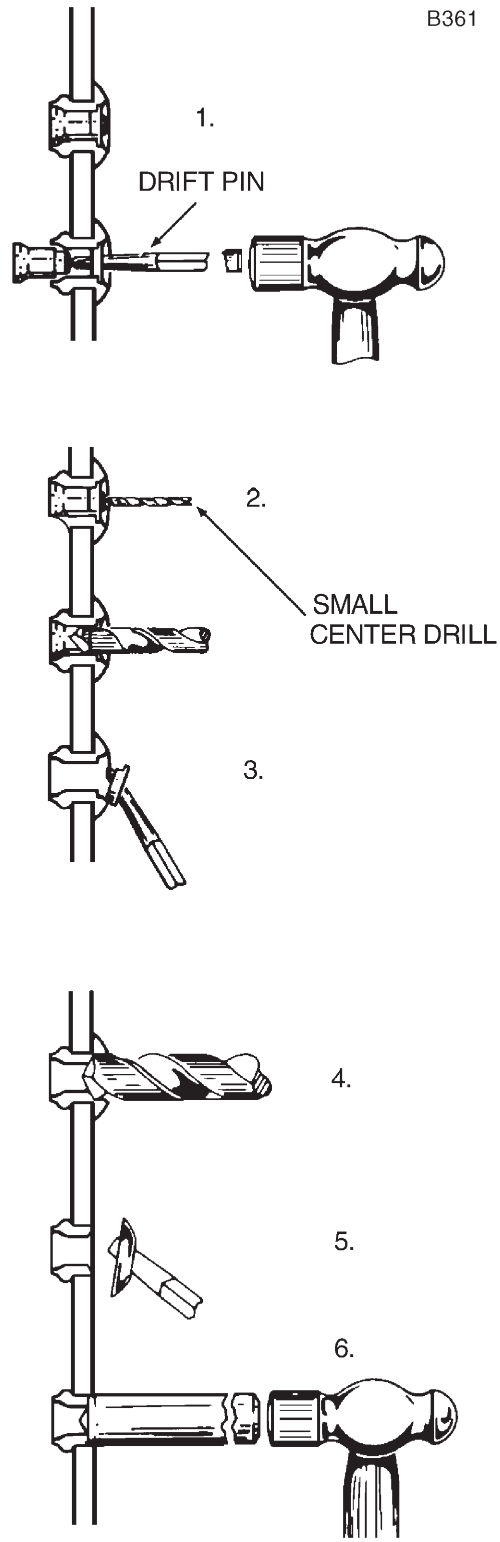
B. To remove from thin material, drill away tapered portion of stem to destroy the lock. Use a small center drill bit on top of the rivet stem to provide a guide for a larger bit. (See "Figure 3", Views 2 and 3.)

C. Pry remainder of locking collar out of rivet head with a drift pin. (See "Figure 3", View 3.)

D. Drill almost, but not completely, through head of rivet. Use a drill bit the same size as the rivet shank. (See "Figure 3", View 4.)

E. Use a drift pin as a lever to break off rivet head. (See "Figure 3", View 5.)

F. Drive out remaining rivet shank with a pin having same diameter as rivet shank. (See"Figure 3", View 6.)



Removing Cherrylock Rivets
Figure 3

5. Identification of Fluid Lines

Refer to “Figure 4”.

Aircraft fluid lines are identified by color code markers, words and geometric symbols. The markers identify each line`s function, content, primary hazard, and the direction of fluid flow.

Most fluid lines are marked with 1 inch tape or decals. Paint is used on lines in the engine induction system.

Certain lines may also be identified as to the specific function within a system. For example: DRAIN, VENT, PRESSURE or RETURN.

Lines conveying fuel may be marked FLAM. Lines containing toxic materials are marked TOXIC. Line containing physically dangerous materials, such as oxygen, nitrogen, or freon, are marked PHDAN.

The aircraft and engine manufacturer is responsible for the original installation of identification markers, Aircraft maintenance personnel are responsible for their replacement when it becomes necessary.

Tapes, paint, tags and decals are placed on both ends of a line and at least once in each compartment through which the line runs. Identification markers are also placed immediately adjacent to each valve, regulator, filter or other accessory within a line.

6. Inspection of Flexible Hoses

NOTE: During the manufacturing process, a condition known as “rubber strike-through” occasionally occurs. This condition is such that rubber material protrudes through the wire braid cover. This condition has no effect on hose quality.

It is recommended that flexible hoses be inspected every 100 hours, especially those in the engine compartments. When inspecting hoses, look for the following conditions:

A. Check each installation to be sure the hose is not kinked, twisted, or distorted. Check for evidence of abrasion, cuts, and broken wires. Random broken wires are acceptable since wire breaks sometimes occur during manufacture. Discard hose if two or more broken wires are found per plait (braid) or more than six broken wires per lineal foot. Broken wires in an area where kinking is evident is also a cause for rejection.

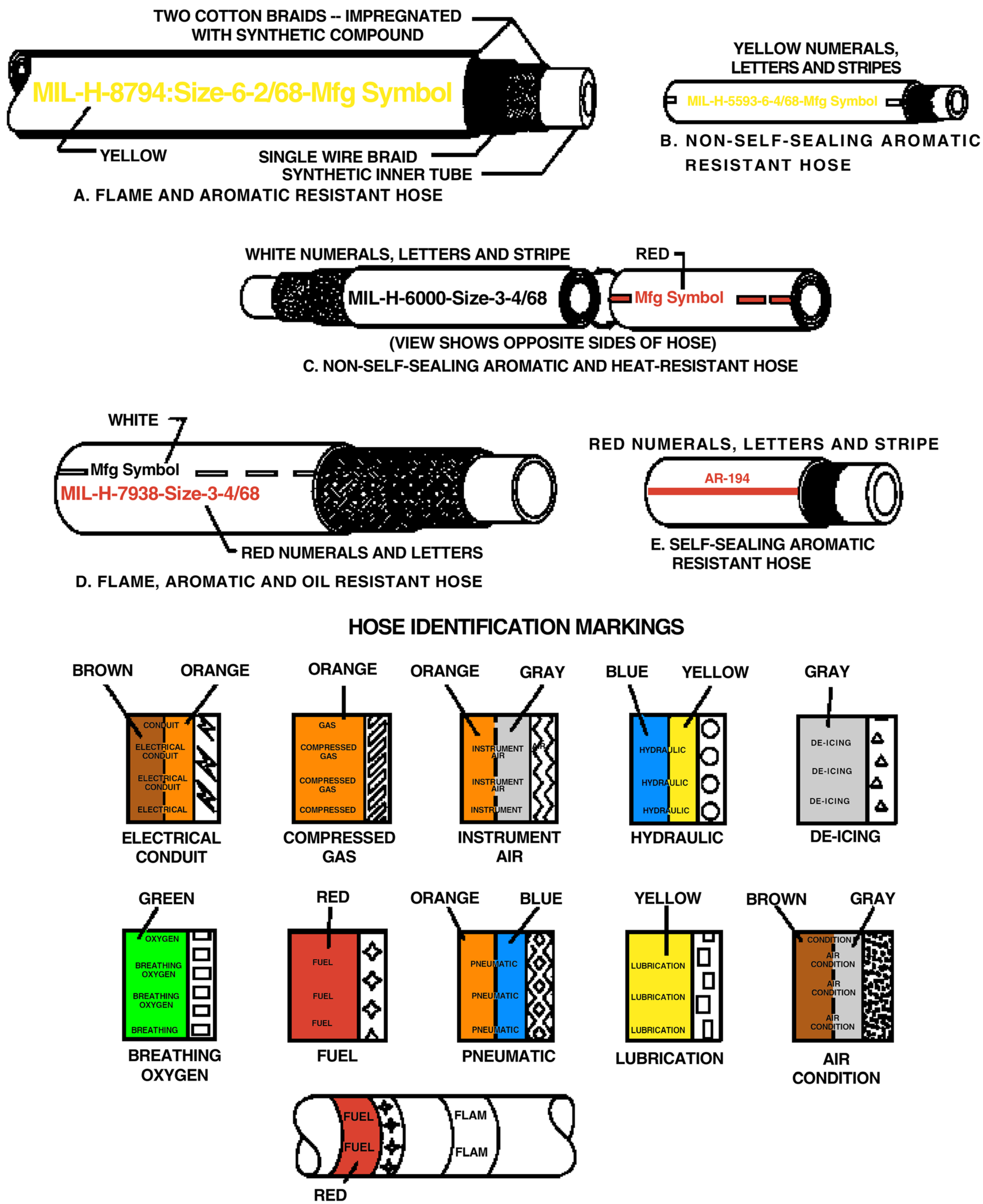
CAUTION: PUNCTURING THE OUTER COVER OF THE HOSE MAY CAUSE DAMAGE TO THE HOSE.

B. Check each assembly for deterioration, ply separation of cover or braid, cracks, weather checking, lack of flexibility, blisters or bulging, collapse, or sharp bending. Blisters on the outer synthetic cover do not necessarily indicate a faulty hose.

C. Remove hose from assembly if hose shows any visible wear. Inspect hose interior and check for signs of deterioration, tube collapse, cut rubber, wire braid puncture, or restriction. To inspect hoses with elbow fittings, use flexible inspection light and viewer, or inspection ball as described in “Chart 1”. Replace hose if any deterioration exists.

CHART 1
BALL DIAMETERS FOR TESTING HOSE RESTRICTIONS

Hose Size	Ball Size
- 4	5/64
- 5	9/65
- 6	13/64
- 8	9/32
- 10	3/8
- 12	1/2
- 16	47/64
- 20	61/64



Hose, Tube, and Line Markings
Figure 4

7. Flared Tube Assemblies

Refer to "Figure 5".

When any fitting containing conical seals is disconnected the following steps must be performed when connecting the fitting.

A. Lubricate threads of nut with a substance compatible with the fluid that is to be contained in the line

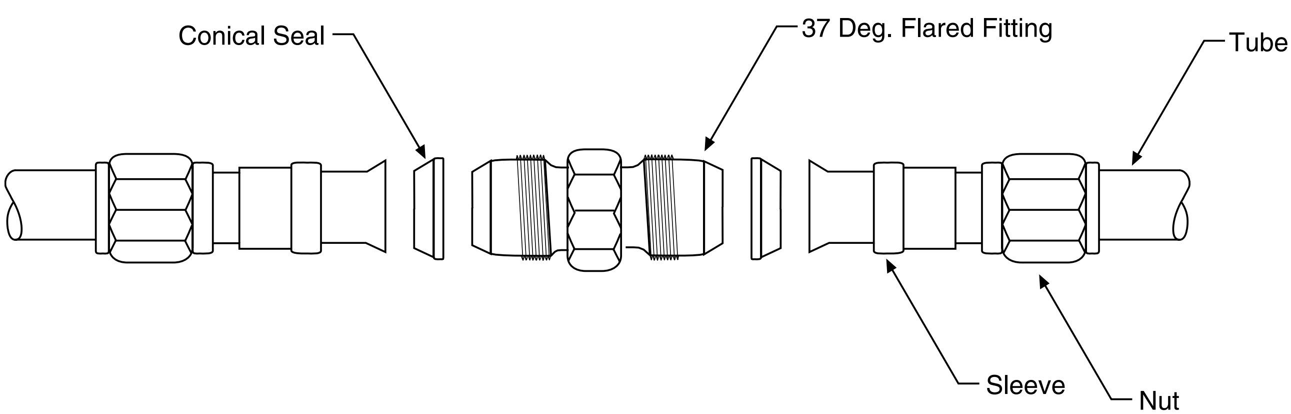
NOTE: Conical seals are never to be reused. If a connection is broken the seals must be replaced.

B. Install new conical seals

CAUTION: ANGULAR MISALIGNMENT BETWEEN TUBE AND FITTING WILL OVER-STRAIN SLEEVE AND NUT DURING TORQUE-UP. APPLY TORQUE ONLY AFTER TUBE CENTER LINE IS SQUARELY MATED TO THE FITTING CENTER LINE.

NOTE: Apply torque on back portion of nut next to shoulder to minimize distortion of thread.

C. Tighten fitting one (1) to one and one half (1 1/2) hex flats from position of sharp torque rise (finger tight).



Conical Seals
Figure 5

8. Flareless Tube Assemblies

The use of flareless tube fittings eliminates all tube flaring. An operation, referred to as presetting, is necessary prior to installing a new flareless tube assembly. Presetting is performed as follows:

A. Cut tube to correct length. Ensure ends are perfectly square. Deburr inside and outside of tube. Slip nut, then sleeve, over the tube. (Refer to “Figure 6”, Step 1.)

B. Lubricate fitting and nut threads as specified in table contained in “Figure 6”.

C. Place fitting in a vise (refer to “Figure 6”, Step 2). Hold tubing firmly and squarely on seat in fitting. (Tube must bottom firmly in the fitting.) Tighten nut until cutting edge of sleeve grips tube. This point is determined by slowly turning tube back and forth while tightening nut. When tube no longer turns, nut is ready for final tightening.

D. Final tightening depends upon type and size of tubing. On aluminum alloy tubing up to and including half inch outside diameter, tighten nut from 1 to 1-1/6 turns. On aluminum alloy tubing over half inch outside diameter, or steel tubing, tighten nut from 1-1/6 to 1-1/2 turns.

E. After presetting the sleeve, disconnect tubing from fitting and check the following points (refer to Figure 6, Step 3):

(1) Tube extends 3/32 to 1/8 inch beyond sleeve pilot to prevent blow off.

(2) Sleeve pilot contacts tube. A maximum clearance of 0.005 inch for aluminum alloy tubing, or 0.015 inch for steel tubing, is acceptable.

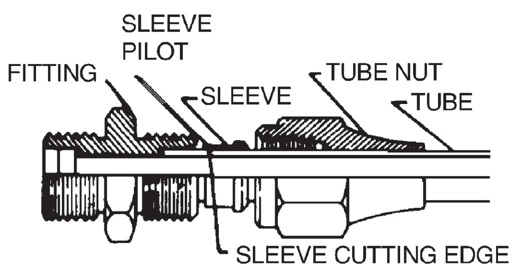
(3) A slight collapse of tube at sleeve cut is permissible. No movement of sleeve pilot, except rotation, is permissible.

B363

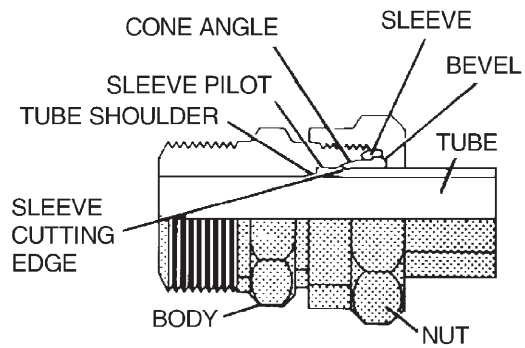
TUBING AND HOSE LUBRICANTS

TUBING SYSTEM	LUBRICANT
HYDRAULIC	MIL-H-5606
FUEL	MIL-H-5656
OIL	SYSTEM OIL
PNEUMATIC	MIL-L-4343
OXYGEN *	NONE

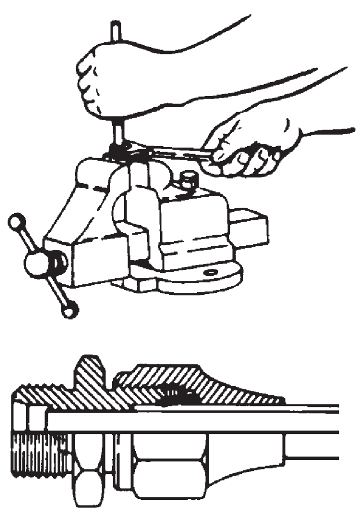
* SEE 35-10-00.



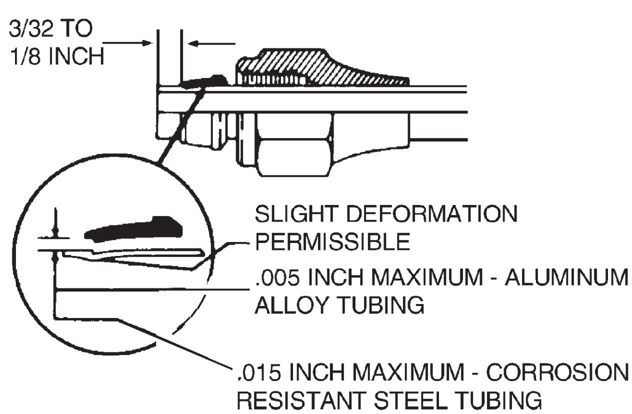
STEP 1



FLARELESS-TUBE FITTING



STEP 2



STEP 3

Flareless Tube Fittings
Figure 6

9. Support Clamps

CAUTION: MAKE CERTAIN THAT CLAMPS ARE OF THE CORRECT SIZE. CLAMPS OR SUPPORTING CLIPS SMALLER THAN THE OUTSIDE DIAMETER OF THE HOSE MAY RESTRICT THE FLOW OF FLUID THROUGH THE HOSE.

Support clamps are used to secure the various lines to the airframe or power plant assemblies. Several type of support clamps are used for this purpose. The rubber cushioned and plain are the most commonly used clamps. The rubber cushioned clamp is used to secure lines subject to vibration; the cushioning prevents chafing of the tubing. The plain clamp is used to secure lines in areas not subject to vibration.

A teflon cushioned clamp is used in areas where the deteriorating effects of hydraulic fluid or fuel is expected, however, because it is less resilient, it does not provide as good a vibration damping effect as other cushion materials.

Use bonded clamps to secure metal hydraulic, fuel and oil lines in place. Unbonded clamps should be used only for securing wiring. Remove any paint or anodizing from the portion of the tube at the bonding clap location.

All plumbing lines must be secured at specified intervals. The maximum distance between supports for rigid fluid tubing is shown in “Chart 2”.

CHART 2
MAXIMUM DISTANCE BETWEEN SUPPORTS FOR FLUID TUBING

	Distance Between Supports (IN.)	
Tube O.D. (IN.)	**Aluminum Alloy**	**Steel**
1/8	9-1/2	11-1/2
3/16	12	14
1/4	13-1/2	16
5/16	15	18
3/8	16-1/2	20
1/2	19	23
5/8	22	25-1/2
3/4	24	27-1/2
1	26-1/2	30

10. Dye Penetrant Inspections

NOTE: The following procedure is general in nature. See manufacturer's instructions, included with dye penetrant kit, for specifics.

A. Using a volatile cleaner, thoroughly remove dirt, loose scale, oil and grease from surface to be inspected.

B. Heat surface to at least 70°F (21°C), but not exceeding 130°F (54°C).

C. Apply penetrant by brushing, spraying, or dipping. Let stand 2 to 15 minutes, depending on temperature.

D. Remove surplus penetrant by applying special cleaner recommended by penetrant manufacturer, or by rinsing with water. Allow housing to dry.

E. Apply a light, even coat of developer by spraying, brushing, or dipping. Cracks or other opening in surface being inspected will appear as bright red. An indication of size of the defect may be obtained by watching the size and rate of growth of red indication.

11. Threaded Fastener Installation

(PIR-PPS20015-1, Rev. AM)

A. General Requirements

Apply the following unless otherwise specified in the subject Chapter or Section:

(1) Fasteners installed in a vertical position must be installed with the fasteners head upward.

(2) Fasteners installed in a horizontal position must be installed with the fasteners head forward.

(3) Fasteners installed in a horizontal inboard-outboard position must be installed with the fastener head inboard.

B. Exceptions

The following exceptions to general installation practices are permissible:

(1) Fasteners installed on the aircraft exterior must be installed with the fasteners head on the outside.

(2) Fasteners which pierce the sides or cover of junction boxes, except the mounting side, must be installed with the head on the outside of the box. The length and location of the bolts must be such as to eliminate any damage to electrical wiring or equipment.

(3) Fasteners used for the final mounting of junction boxes must be installed with the fasteners head on the inside of the box.

(4) All fasteners must be installed in such a manner as to minimize injury to personnel or other damage (torn clothing, etc.).

(5) Fasteners centered across butt line zero or fasteners used for non-structural purposes (e.g. securing of electrical or hydraulic line), may be installed in any position to suit installation convenience unless specifically noted otherwise on the drawing.

C. Thread Lubrication

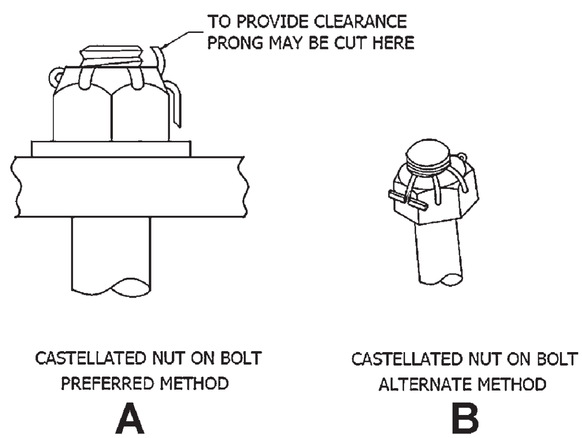
Lubricants other than that supplied on the fastener as purchased must not be used unless specified in the subject Chapter or Section. See "Thread Lubrication" on page 200023 for general requirements.

D. Cotter Pin Installation

(PIR-PPS20023, Rev. F.)

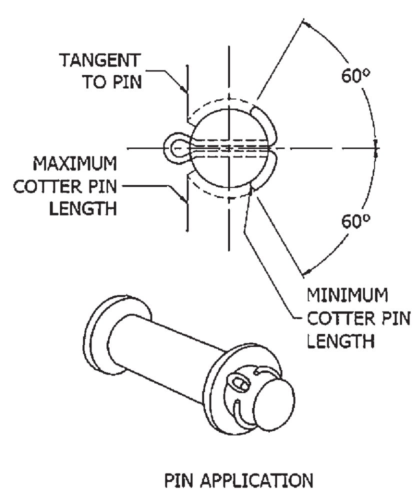
(1) Installation of cotter pins in castellated nuts must conform to methods shown in “Figure 7” View A or B. In either case, the pin must be installed with the head parallel to the slot in the nut.

(2) Cotter pins in castellated nuts must not extend more than 50% of the cotter diameter above the top of the slot as indicated in “Figure 7” View A. Prior to installation of cotter pin, tighten nut to the high side of the selected torque range, and if necessary, back off until the closest slot aligns with the hole. If axial hole alignment is not possible within the necessary grip range, change/add washers per “Washers” on page 200012 until hole alignment is possible.



Cotter Pin Installation – Castellated Nut
Figure 7

(3) Cotter pins must be installed in clevis pins as shown in “Figure 8” with the axis of the eye parallel to the shank of clevis pin or rod end and bend the prongs around the shank of the pin or rod end.



Cotter Pin Installation – Clevis Pin
Figure 8

(4) Largest nominal diameter cotter pin which the hole and slots will accommodate must be used, but in no application to a nut, bolt, or screw must pin size be less than the sizes listed in “Chart 3”.

CHART 3
COTTER PIN SIZE

Thread Size (Inches)	Minimum Pin Size (Inches)
0.1380	0.028
0.1640	0.044
0.1900	
0.2500	
0.3125	
0.3750	0.072
0.4375	
0.5000	
0.5625	0.086
0.6250	
0.7500	
0.8750	
1.0000	
1.1250	0.116
1.2500	
1.3750	
1.5000	

E. Fastener Lengths

Fastener shank lengths must be long enough to prevent bearing loads on threads. In addition, the fastener or screw must extend through the nut. A variation of specified fastener grip length by one size (longer or shorter) to meet requirements is allowed.

F. Washers

(1) A total of three washers is permitted, one for surface protection and two for grip adjustment.

(2) A maximum of two NAS1149 (filler) washers of the correct diameter, material, and finish that matches fastener being installed may be added under fastener heads or nuts to correct for variations in material thickness within the tolerances permitted.

(3) A maximum of two standard filler washers may be used under the nut where needed to adjust fastener length or alignment of cotter key hole.

(4) A maximum of two standard filler washers may be used under the fastener head to adjust for protruding head fastener length where nutplates are used.

(5) One standard filler washer may be used under the nut or fastener head (whichever is being turned) to protect the surface from damage while tightening.

(6) Where filler washers are required in conjunction with insulating washers, the filler washer must be between the insulating washer and the bolt head or nut.

(7) Where lock washers are specified on Engineering drawings to be used in contact with aluminum parts, a standard filler washer must be used between the lock washer and surface of the part prevent damage.

G. Self-locking Fasteners

Limitations of the use of self-locking nuts, fasteners, and screws, including fasteners with non-metallic inserts are as follows:

(1) Fasteners incorporating self-locking devices must not be re-used if they can be run up using less than the required minimum torque values specified in "Torque Requirements" on page 200015. They may be re-used if hand tools are required to run them up, providing there is no obvious damage to the self-locking device prior to installation.
(2) Fasteners 0.3125 (5/16) inch diameter and over with cotter pin holes may be used with self-locking nuts.
(3) Self-locking nuts (non-metallic) must not be used on drilled bolts, or studs, 0.25 (1/4) inches or less in diameter.
(4) Self-locking nuts and fasteners must not be used at joints which subject either the nut or the fastener to rotation. Only self-locking nuts may be used with anti-friction bearings and control pulleys provided the inner face of the bearing is clamped to the supporting structures by the nut and the fastener.
(5) Corrosion-resisting steel self-locking nuts must be used only with corrosion-resisting steel bolts.
(6) Self-locking fasteners must never be tapped or re-threaded. Nuts, fasteners, and screws with damaged threads or rough ends must not be used or re-threaded.
(7) Self-locking fasteners must be used only in applications which permit engagement of at least three threads past the locking device with simultaneous engagement of the locking device.
(8) Self-locking fasteners must not be used in conjunction with those applications requiring controlled torques, (i.e., molded rubber gaskets in fuel cells, etc.).
(9) Self-locking fasteners that have had the locking element reworked or reprocessed by other than an approved bolt manufacturer must not be used.
(10) Non-metallic insert type self-locking bolts are intended for use at temperature conditions ranging from -65°F to 250°F and are designed to function satisfactorily through that range.
(11) Plating or re-plating of non-metallic (fiber or nylon insert) self-locking nuts is not permitted because of the deterioration of the insert by the plating solution and post plating baking.

H. Multiple Fastener Installations

Multiple fastener installations must be subject to the following steps unless otherwise specified in the subject Chapter or Section:

(1) Finger tighten all nuts first.
(2) Using appropriate wrench, snug up all the nuts.
(3) Torque opposite nuts or fasteners. Do not successively tighten adjacent nuts.

I. Fastener Head Shaving

Fastener heads must not be shaved unless specified in the subject Chapter or Section.

J. Inspection

(1) Cracks in Fastener Heads

Any bolt or bolt installation where open cracks are visible in the bolt head requires replacement of the bolt.

(2) Gaps Under Fastener Heads

The presence of gaps between the fastener head and the bearing surface, which allows for a 0.004 inch maximum feeler gage to be inserted, is allowable providing that not more than 40 percent of the fastener head circumference has this gap.

(3) Torque Seal

The application of torque seal is used to indicate that threaded fasteners have been properly torqued. Unless otherwise specified, torque seal must not be applied to fasteners used in the following installations:

(a) Where normal vibration may cause seal to fall off fasteners into openings in electronic devices.

(b) Fasteners having direct contact with fuel, hydraulic, or de-ice fluids.

(c) Inside ducts and manifolds of induction, heating, and ventilation systems.

12. Torque Requirements

CAUTION: DO NOT OVERTORQUE FITTINGS.

"Chart 4" lists the torque values for flared fittings of various sizes and material.

NOTE: When installing flared fittings, verify that male threads are properly lubricated.

The torque values given in "Chart 5" on page 200018 are derived from oil-free cadmium-plated threads and are recommended for all airframe installation procedures where torquing is required, unless other values are specified in subject Chapter or Section. Engine torque values are found in the latest revision of Lycoming Service Table of Limits SSP-1776 and propeller torque values are found in 61-10-00.

NOTE: All torque values given are installation torques for installation purposes only and must not be construed as retained torque.

NOTE: If normal operation requires movement between any of the components being clamped together, tighten the nut (or bolt) enough to ensure intended operation of the assembly.

A. Calibrate the torque wrench periodically to assure accuracy, and recheck frequently. See "Torque Wrenches" on page 20001.

B. If the fastener, screw, or nut is listed in "Chart 5" on page 200018 but the mating fastener is not listed, tighten only to the low end of the torque range specified for the listed fastener.

CHART 4
FLARE FITTING TORQUE VALUES

(PIR-AC65-9A.)

Torque — Inch-Pounds						
Tubing OD Inches	**Aluminum Alloy Tubing**		**Steel Tubing**		**Hose End Fitting and Hose Assemblies**	
	Minimum	**Maximum**	**Minimum**	**Maximum**	**Minimum**	**Maximum**
1/8	20	30	———	———	———	———
3/16	30	40	90	100	70	120
1/4	40	65	135	150	100	250
5/16	60	85	180	200	210	420
3/8	75	125	270	300	300	480
1/2	150	250	450	500	500	850
5/8	200	350	650	700	700	1150
3/4	300	500	900	1000	———	———
7/8	500	600	1000	1100	———	———
1	500	700	1200	1400	———	———
1-1/4	600	900	———	———	———	———
1-1/2	600	900	———	———	———	———

C. Unless otherwise specified in the subject Chapter or Section or as specified by the engine manufacturer or as specified in “AN386 Taper Pin Installation” on page 200017, threaded fasteners must be torqued to final torque values as determined by adding friction drag to the torque values listed in Tables 3–6 in “Chart 5” on page 200018. The final torque values (with friction drag included) are shown in Tables 3–6 for some select fastener/nut combinations. If the final torque value is not listed, then it must be calculated per paragraphs (2) and (3). In addition, the following limitations and exceptions apply:

(1) Fastener and nut threads must be clean and dry (free of lubricants). If the subject Chapter or Section requires the fastener and/or nut to be lubricated prior to tightening, but does not specify a torque requirement, the final torque range (including friction drag torque) must be 50 percent of the value shown in Tables 3–6.

(2) The friction drag torque can be determined as follows:

Run the nut down to near the contact (but not in contact) with the bearing surface. While still not contacting the bearing surface, the “friction drag torque” must be determined by observing the reading on face of “dial type” wrenches or observing the nearest detent adjustment setting for “snap-over” wrenches.

(3) The friction drag (if any) must be added to the desired torque specified by Tables 3–6. This is referred to as “final torque”, which should register on the indicator or be the setting for a snap-over wrench. The final torque values (with friction drag included) are shown in Tables 3–6 for some select fastener/nut combinations. If the value is not listed, then it must be calculated per paragraph (2).

(4) If the friction drag of the nut is significantly low, such that it cannot be accurately measured (free running nuts), then adding friction drag to determine final torque is not required.

(5) Torque requirements do not apply to cross recessed, slotted screws, or fasteners installed into rivnuts, nor press nuts or other nuts that are not designed to rotate for wrenching at the tail end unless otherwise specified by the applicable drawing.

(6) Fasteners listed in Tables 3–6, installed into nut plates and are accessible to be torqued at the fastener, must be tightened to the low end of the torque range found for shear applications listed in tables Tables 3–6 unless otherwise specified in the subject Chapter or Section.

(7) When the fastener is stationary and the nut is torqued, use the lower side of the torque range. When the nut is stationary and the fastener is torqued, use the higher side of the torque range. When the nut is stationary and the fastener is being torqued, ensure one (1) washer is installed under the head as follows:

(a) If the subject Chapter or Section does not specify the use of a washer under the head, install one (1) NAS1149 0.032 inch thick washer under the head. If additional washers are required under the nut to adjust for grip length variation per “Washers” on page 200012, reduce them to 0.032 inches to allow for the additional 0.032 inch thick washer now installed under the head. Check to ensure threads are not bearing per “Fastener Lengths” on page 200012, due to the added 0.032 inch washer thickness.

(b) All added washers are to be of the correct diameter, material, and finish that match the drawing type washer being installed.

(8) Apply a smooth even pull when applying torque pressure. If chattering or a jerking motion occurs during final torque, back off and re-torque.

(9) For castellated nuts, torque nut onto the fastener joint to the maximum allowable torque for the particular fastener/nut combination then back off to the nearest castellation, and install the cotter pin. Unless otherwise specified in the subject Chapter or Section, when castellated nuts are used with a cotter pin on moving joints, nuts must be tightened adequately as to remove looseness in the joint and allow for cotter pin installation.

(10) Unless otherwise specified specified in the subject Chapter or Section, when parts are used on engines or other supplied assemblies using Piper furnished or existing supplier threaded fasteners, tightening torque limits shall be as specified by the latest applicable supplier specifications.

(11) Unless otherwise specified in the subject Chapter or Section, the following screws will not require torqueing to a specific value. They only require torqueing until the fastener is fully seated and there is no play or looseness in the joint using hand pressure. Fasteners in this category are MS35206, MS35207, MS24693, MS35190, and MS35191. Torque striping these fasteners is not required.

D. Marking After Tightening

Nuts (or fasteners and screws if no nut is involved) that have been final tightened to torque values as specified per paragraph (10) on page 200016 and Tables 3–6 in "Chart 5" on page 200018, or as specified in the subject Chapter or Section, must be marked with torque seal.

E. AN386 Taper Pin Installation

(1) If not already accomplished:

(a) Enlarge pilot holes with standard drill until an appropriate Brown & Sharpe tapered reamer (or equivalent) can be used.

(b) Ream the hole slowly, periodically inserting the specified taper pin as a guide, until the small end of the taper pin protrudes no more than 0.063 inches past the hole and the large end protrudes between 0.125 and 0.250 inches per NASM386. If the large end protrusion does not meet the requirements, adjust the grip length of the pin accordingly.

NOTE: Ream slowly, making sure to keep the axis of the reamer aligned as close as possible to the axis of the hole.

(2) Unless otherwise specified in the subject Chapter or Section, assemble the AN386 pin, AN975 washer, and castle nut as specified in the subject Chapter or Section, and torque nut per paragraph (8) on page 200016 until the cotter pin may be inserted.

F. Gap Conditions Between Parts Attached with Threaded Fasteners

If a gap condition exists between mating parts where a threaded fastener is to be installed, install the required fasteners and associated hardware except do not torque to final requirement. Instead, torque to a value 10% of the maximum required torque (from Tables 3–6, not including friction drag), and add the friction drag of the required (installed) nut per paragraph (2) on page 200016. If the final torque (including friction drag) is listed in Tables 3–6, the gap-closing torque may be calculated as follows:

$$T_{MAX} = T_D - (0.90 \text{ X } T_{ND})$$

Variable Definitions:

T_{MAX}: Permissible gap-closing torque

T_D: Final torque including friction drag (Tables 3–6 in "Chart 5" on page 200018)

T_{ND}: Final torque not including friction drag (Tables 3–6 in "Chart 5" on page 200018)

Review the example below referencing the AN6 fastener (3/8-24) from Table 3:

$$T_{MAX} = T_D - (0.90 \text{ X } T_{ND})$$
$$T_{MAX} = 270 \text{ in·lbs} - (0.90 \text{ X } 190 \text{ in·lbs})$$
$$T_{MAX} = 99 \text{ in·lbs}$$

NOTE: If the torque requirement listed in Tables 3 thru 6 exceeds the requirement specified in the subject Chapter or Section, the requirement specified in the subject Chapter or Section must be used to determine the permissible torque. Accomplish this for all fasteners common to gapped interface. If no gap exists after torqueing per the above procedure, finish torqueing to the final requirement. If a gap remains, consult your Piper Dealer's Service Advisor (DSA) for further assistance.

CHART 5 (Sheet 1 of 4)
TORQUE TABLES

(1) The following tables are partially derived from FAA AC 43.13-1B.

(2) Some fasteners and nuts are listed in more than one group. Select torque from appropriate group depending on application (tension vs shear).

(3) If the chapter/section calls out a fastener and nut that are not in the same group, use the lower of the two torque values.

(4) NASM21042 contains wrenching torque values that are intended for testing purposes only, for aircraft assemblies use torque values shown herein.

(5) Although MS21042, NAS1291, and NAS679 nuts are reduced height, they are included in both tension and shear groups and may be used for both applications.

(6) Unless otherwise specified in the subject Chapter or Section, size #6 screws used with self-locking nutplates shall be torqued to no greater than 4–5 in·lbs with an appropriate calibrated driver.

(7) If final torque (with friction drag included) is not listed it shall be calculated per paragraphs (2) on page 200016 and (3) on page 200016.

NOTE: Within Chart 5, Tables 1 and 2 are not used.

CHART 5 (Sheet 2 of 4)
TORQUE TABLES

Table 3: Standard Fasteners

Standard Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in·lbs)		Torque (in·lbs)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
AN3 thru 20 AN42 thru 49 AN386 AN525 MS20033 thru 20046 MS20073 MS20074 MS21046 MS24694 MS27039	Tension AN310 AN315 MS20365[1] MS21042 MS21044 MS21045 NAS679 NAS1291 Shear AN316 AN320 MS20364[1] MS21083 MS21245 NAS509	Fine Thread	8-36	12 –15		7 – 9	
			10-32	20 – 25	38 – 43	12 – 15	30 – 33
			1/4-28	50 – 70	65 – 100	30 – 40	60 – 70
			5/16-24	100 – 140	160 – 200	60 – 85	120 – 145
			3/8-24	160 – 190	240 – 270	95 – 110	175 – 190
			7/16-20	450 – 500	550 – 600	270 – 300	370 – 400
			1/2-20	480 – 690	-	290 – 410	-
			9/16-18	800 – 1000	-	480 – 600	-
			5/8-18	1100 – 1300	-	600 – 780	-
			3/4-16	2300 – 2500	-	1300 – 1500	-
			7/8-14	2500 – 3000	-	1500 – 1800	-
			1-14	3700 – 4500	-	2200 – 3300	-
			1 1/8-12	5000 – 7000	-	3000 – 4200	-
			1 1/4-12	9000 - 11000	-	5400 - 6600	-
		Coarse Thread	8-32	12 – 15	27 – 30	7 – 9	22 – 24
			10-24	20 – 25	38 – 43	12 – 15	30 – 33
			1/4-20	40 – 50	70 – 80	25 – 30	55 – 60
			5/16-18	80 – 90	140 – 150	48 – 55	108 – 115
			3/8-16	160 – 185	240 – 265	95 – 110	175 – 190
			7/16-14	235 – 255	335 – 355	140 – 155	240 – 255
			1/2-13	400 – 480	-	240 – 290	-
			9/16-12	500 – 700	-	300 – 420	-
			5/8-11	700 – 900	-	420 – 540	-
			3/4-10	1150 – 1600	-	700 – 950	-
			7/8-9	2200 – 3000	-	1300 – 1800	-
			1-8	3700 – 5000	-	2200 – 3000	-
			1 1/8-8	5500 – 6500	-	3300 – 4000	-
			1 1/4-8	6500 – 8000	-	4000 – 5000	-

(1) Superceded by MS21042.

CHART 5 (Sheet 3 of 4)
TORQUE TABLES

Table 4: High Strength Fasteners

High Strength Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in·lbs)		Torque (in·lbs)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
MS20004 thru 20024 NAS333 thru 340 NAS464 NAS624 thru 644 NAS1580 NAS6203 thru 6220 NAS6303 thru 6320[2] NAS6603 thru 6620 NAS6703 thru 6720	<u>Tension</u> AN310 AN315 MS20365[1] MS21042 MS21044 MS21045 NAS679 NAS1291 <u>Shear</u> AN316 AN320 MS20364[1] MS21083 MS21245 NAS509	Fine Thread	8-36	No Data	-	No Data	-
			10-32	25 – 30	46 – 48	15 – 20	33 – 38
			1/4-28	80 – 100	110 – 130	50 – 60	50 – 90
			5/16-24	120 – 145	180 – 205	70 – 90	130 – 150
			3/8-24	200 – 250	280 – 330	120 – 150	200 – 230
			7/16-20	520 – 630	620 – 730	300 – 400	400 – 500
			1/2-20	770 – 950	-	450 – 550	-
			9/16-18	1100 – 1300	-	650 – 800	-
			5/8-18	1250 – 1550	-	750 – 950	-
			3/4-16	2650 – 3200	-	1600 – 1900	-
			7/8-14	3550 – 4350	-	2100 – 2600	-
			1-14	4500 – 5500	-	2700 – 3300	-
			1 1/8-12	6000 – 7300	-	3600 – 4400	-
			1 1/4-12	11000 – 13400	-	6600 – 8000	-
		Coarse Thread	All	No Data			

(1) Superceded by MS21042.
(2) Not applicable for aluminum coated fastebers.

CHART 5 (Sheet 4 of 4)
TORQUE TABLES

Table 5: Aluminum Fasteners

Aluminum Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in·lbs)		Torque (in·lbs)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
AN3DD	Tension: AN310D, AN315D; Shear: AN316D, AN320D	Fine Thread	8-36	5 – 10	-	3 – 6	-
			10-32	10 – 15	28 – 33	5 – 10	23 – 28
			1/4-28	30 – 45	60 – 75	15 – 30	45 – 60
			5/16-24	40 – 65	100 – 125	25 – 40	85 – 100
			3/8-24	75 – 110	155 – 190	45 – 70	125 – 150
			7/16-20	180 – 280	280 – 380	110 – 170	210 – 270
			1/2-20	280 – 410	-	160 – 260	-
		Coarse Thread	All	No Data			

Table 6: MS17825 Castle Nuts

Aluminum Fasteners		Fastener Size		Tension		Shear	
Fasteners	Nuts			Torque (in·lbs)		Torque (in·lbs)	
				No Friction Drag	W/ Friction Drag	No Friction Drag	W/ Friction Drag
.190 .250 .3125 .375 .4375 .500	Tension: MS17825; Shear: MS17826	Fine Thread	8-36	No Data	N/A	No Data	-
			10-32	28		16	-
			1/4-28	65		35	-
			5/16-24	180		70	-
			3/8-24	260		100	-
			7/16-20	460		180	-
			1/2-20	720		240	-
			9/16-18	880		320	-
			5/8-18	1300		480	-
			3/4-16	2200		880	-
			7/8-14	3700		1500	-
			1-14	5400		2400	-
			1 1/8-12	8000		4000	-
			1 1/4-12	11000		5600	-
		Coarse Thread	All	No Data			

NOTE: See paragraph (9) on page 200016 for castellated nut installation requirements.

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13. Thread Lubrication

Lubricate all fittings on external lines, including attachment points at engine and other components, with proper lubricant as specified in “Chart 6”.

When applying thread lubricants, proceed as follows:

A. Thoroughly clean threads before applying lubricant.

B. Use thread lubricant sparingly.

C. Apply thread lubricant to male threads only.

D. Lubricate first three threads only on straight fittings.

E. Do not lubricate first two threads on tapered fittings; apply lubricant to next three threads only.

F. Ensure lubricant does not enter fittings or flared areas.

G. Lubricate any fittings going to engine with same fluid going through lines.

CHART 6
THREAD LUBRICANTS

Line	Lubricant
WARNING: DO NOT PERMIT SAE-AMS-2518 ANTI-SEIZE COMPOUND TO ENTER SYSTEM. APPLY TO FITTING THREADS ONLY.	
Air Conditioning Refrigerant	SAE-AMS-2518, Anti-Seize Compound, Graphite Petrolatum
Brakes	MIL-PRF-5606, Hydraulic Fluid
Fuel	SAE-AMS-2518, Anti-Seize Compound, Graphite Petrolatum
Oil	SAE-AMS-G-6032, Grease (Gasoline and Oil Resistant)
Oxygen	A-A-58092, Ribbon Dope Thread Sealant, Anti-Seize and Sealing, Permacel 412
Pitot and Static	TT-A-580 (TT-S-1732), Anti-Seize Compound
Pneumatic Deicer	TT-A-580 (TT-S-1732), Anti-Seize Compound
CAUTION: LUBRICATE ENGINE FITTINGS ONLY WITH THE FLUID CONTAINED IN THE PARTICULAR LINE.	

14. Airframe Water Leak Check

(PIR-PPS60047, Rev. C.)

Use the following procedures to verify watertight integrity of the fuselage.

A. Required Equipment

(1) Choice of one of the following size/length combinations of standard garden hose:

3/4 inch inside diameter, maximum length 150 feet, or

5/8 inch inside diameter, maximum length 75 feet, or

1/2 inch inside diameter, maximum length 25 feet.

(2) A hose nozzle capable of adjustment. (Green Garden Nozzle 1220-C or equivalent is acceptable.)

B. Procedure

(1) Attach hose with nozzle to water tap at city water pressure. Check to insure that there are no leaks in the hose or fittings which would impair the check or cause other damage, with the tap full open.

(2) The check (except for baggage compartments located apart from the cabin) shall be conducted with an observer inside the aircraft.

(3) All cabin and baggage compartment doors and other controllable external closures shall be closed and latched during rainfall simulation.

(4) The spray shall be aimed directly at the openings around doors and windows, and at any other cockpit/cabin/separate baggage compartment closures, for a period of one-half to one and one-half minutes for each door, window, and other closure(s). The nozzle shall be held three to ten feet from point of water contact.

(5) If a leak allowing water to enter the cockpit/cabin/separate baggage compartment is detected, the sealant or caulking (see 91-10-00, Chart 10) shall be repaired, as required, and the check shall be repeated in the area where the leak was noted.

PAINTING

1. Painting Safety

WARNING: OVERSPRAY FROM CERTAIN ENAMELS, IF PUT IN WATER, IS FLAMMABLE. STORE ALL OVERSPRAY IN COVERED CONTAINERS AWAY FROM BUILDINGS WHERE SPRAYING OPERATIONS ARE CONDUCTED.

WARNING: WASH ALL RAGS AND SPONGES USED TO APPLY ANY PHOSPHORIC ACID CONVERSION COATINGS (ALODINE) BEFORE DISPOSAL. IF MATERIAL DRIES ON RAG, THERE IS DANGER OF SPONTANEOUS COMBUSTION.

WARNING: MIX DOPES AND LACQUERS WITH AIR DRILL. DO NOT USE ELECTRIC DRILL. ARCING ELECTRIC DRILL MOTOR WILL IGNITE FUMES.

WARNING: VERIFY SPRAY ROOM IS WELL VENTILATED. A CONCENTRATION OF FUMES WILL CAUSE A DANGEROUS FIRE HAZARD OR INSUFFICIENT OXYGEN FOR THE OPERATOR.

CAUTION: DO NOT ALLOW PAINT STRIPPER TO CONTACT FIBERGLASS REINFORCED PARTS SUCH AS RADOMES, RADIO ANTENNAS, WING PARTS, OR WING TIPS. FIBERGLASS STRUCTURES MAY BE FINISHED WITH ACRYLIC LACQUER OR POLYURETHANE ENAMEL AND ARE DAMAGED BY THE STRIPPER.

2. Polyurethane Paint Safety

WARNING: POLYURETHANE PAINT MAY BE DANGEROUS TO YOUR HEALTH. SERIOUS INJURY WILL RESULT IF SAFETY PRECAUTIONS ARE NOT FOLLOWED.

WARNING: DURING TRANSIT AND STORAGE CHECK FOR SIGNS OF A BULGING CAN, OTHER THAN NORMAL ODOR, OR A CHANGE IN RESIN FROM A CLEAR TO A CLOUDY STATE. A SLOW CARBON DIOXIDE BUILDUP WILL CAUSE CAN TO BURST. REMOVE AND PROPERLY DISPOSE ANY DEFECTIVE CANS.

WARNING: ENSURE ADEQUATE VENTILATION AND WEAR APPROPRIATE BREATHING PROTECTION FACE MASK WHEN PAINTING.

WARNING: POLYURETHANE PAINTS CAN PRODUCE IRRITATION OF THE SKIN, EYES, AND RESPIRATORY TRACT DURING MIXING AND APPLICATION. EXPOSURE TO SPRAY VAPORS AND MISTS DURING SPRAY APPLICATION MAY CAUSE BREATHING DIFFICULTY, SHORTNESS OF BREATH, AND DRY COUGH. INDIVIDUAL SUSCEPTIBILITY IS A CONTROLLING FACTOR. ONCE SENSITIZED, MANY PEOPLE CANNOT TOLERATE ANY EXPOSURE AND MUST THEREAFTER AVOID EXPOSED WORK AREAS.

WARNING: PRODUCTION TYPE MIXING AND SPRAY PAINTING OPERATIONS MUST BE IN SPECIALLY DESIGNED, EXHAUST-VENTILATED AREAS.

WARNING: PAINTERS MUST BE FULLY CLOTHED WITH COLLARS BUTTONED AND SLEEVES TAPED AT THE WRIST. PAINTERS MUST WEAR FITTED, DOUBLE CARTRIDGE ORGANIC VAPOR RESPIRATOR WITH FRESH CARTRIDGE INSERTED DAILY, SOLVENT-RESISTANT GAUNTLET STYLE GLOVES, AND SAFETY GOGGLES.

3. Paint Application

WARNING: GROUND AIRCRAFT BEFORE PAINTING SO NO STATIC ELECTRICITY CHARGES BUILD UP AND DISCHARGE.

CAUTION: PROTECT WINDSHIELD WHEN MASKING AIRCRAFT. PAINT STRIPPERS, METAL BRIGHTENERS, AND SOLVENTS WILL DAMAGE WINDSHIELD.

CAUTION: BALANCE MOVABLE CONTROL SURFACES AFTER PAINTING. REFER TO APPROPRIATE MAINTENANCE MANUAL SECTIONS.

CAUTION: BEFORE FORCE DRYING AT ELEVATED TEMPERATURES, VERIFY THAT ALL FUEL TANK VENTS ARE UNOBSTRUCTED AND WILL NOT RESULT IN EXPANDED FUEL SPILLING ON NEWLY PAINTED SURFACES OR PAINT BOOTH FLOOR.

CAUTION: DO NOT PAINT PITOT TUBES, GAS CAPS, OR ANTENNA COVERS THAT WERE NOT FACTORY PAINTED.

CAUTION: DO NOT USE METALLIC PAINTS ON RADAR CONES OR ANTENNA COVERS.

CAUTION: DO NOT ALLOW SILICONE LUBRICANTS TO CONTACT ANY SURFACES TO BE PAINTED. SILICONE LUBRICANT IS VERY DIFFICULT TO REMOVE COMPLETELY.

4. Painting Sequence

For primer, tack, finish coats, and lacquer application:

A. Position airplane so airflow is from tail toward nose and overspray ahead of you.

B. To minimize overspray problems, have two painters work simultaneously on opposite sides of airplane.

C. Paint difficult areas such as landing gear, and wheel wells before flat surfaces. Paint the ends and leading edges of ailerons and flaps. Paint flap and aileron wells, wing tips, and leading and trailing edges.

D. Paint the bottom of the airplane first including bottom of horizontal tail surfaces. Starting at the root and working outward, spray chordwise. Work up fuselage and allow spray to cover sides. Work up to engine. Spray wing bottom. Start each painter at the root and work toward tip, spraying chordwise.

E. Lower airplane tail enough to reach fin top. When spraying fuselage top, tilt spray gun so overspray is ahead of area being painted and new paint will wipe out overspray. Spray primer across fuselage, vertical and horizontal tail surfaces, and wing.

5. Color Matching

See aircraft logbooks for color codes.

6. Trim and Registration Numbers

Apply predominant color first over entire surface. Apply trim colors over base color after it dries. When top of fuselage is to be painted white with a dark color adjoining it, apply light color and feather into area to be painted with dark color. When light color dries, place masking tape and paper along separation line, and apply dark color.

Allow paint to dry several hours before removing masking tape. Remove tape by pulling slowly parallel to surface. This will reduce the possibility of peeling off finish with tape.

Apply registration numbers by painting or affixing self-adhering plastic figures. They must be solid color lines contrasting with background. Location and size of identification numbers vary, per aircraft size. Location and size is found in Federal Aviation Regulations.

7. Paint System Compatibility

Before applying new paint, find what type finish was used previously. Refer to the Piper parts catalog for correct paint number and color.

Identify paint finishes by applying engine oil to a small surface area. Old nitrocellulose finishes will soften in a few minutes. Acrylics, urethanes, and epoxy finishes show no effects.

If not identified, wipe down a small area with rag wet with methy ethyl ketone. MEK picks up pigments from acrylic finishes, but not from epoxy or cured urethane coatings. Wipe surface, do not rub. Heavy rubbing picks up epoxy and urethane pigments from coatings not fully cured.

The use of different types of paint, with several coatings, make repair of damaged and deteriorated areas difficult. Paint finishes are not always compatible. The following are general rules for compatibility and are not necessarily listed in order of importance.

A. Old type zinc chromate primer may be used directly for touchup of bare metal surfaces and on interior finishes. It may be overcoated with wash primers if in good condition. Acrylic lacquer finishes will not adhere to this material.

B. Modified zinc chromate primer will not adhere to bare metal. Never use it over a dried film of acrylic nitrocellulose lacquer.

C. Nitrocellulose coatings will adhere to acrylic finishes, but reverse is not true. Do not use acrylic nitrocellulose lacquers over old nitrocellulose finishes.

D. Acrylic nitrocellulose lacquers will not adhere to nitrocellulose and epoxy finishes and to bare metal. For best results, apply lacquers over fresh, successive coatings of wash primer and modified zinc chromate. They also adhere to freshly applied epoxy coatings (dried less than 6 hours).

E. Epoxy topcoats adhere to all paint systems in good condition. Use epoxy for general touchup, including touchup of defects in baked enamel coatings.

F. Old wash primer coats may be overcoated directly with epoxy finishes. Apply a new second coat of wash primer if an acrylic finish is to be applied.

G. Old acrylic finishes may be refinished with new acrylic provided old coating is thoroughly softened using acrylic nitrocellulose thinner before paint touchup.

H. Repair damage to epoxy finishes by using more epoxy. Neither lacquer finish will stick to epoxy surfaces. In some instances, air drying enamels may be used for touchup of epoxy coatings if edges of damaged areas are roughened with abrasive paper.

8. Common Paint Troubles

A. Poor Adhesion - Paint properly applied to correctly pretreated surfaces will adhere satisfactorily. When thoroughly dry, paint must not be easily removed. Poor adhesion can result from:

(1) Inadequate cleaning and pretreatment.
(2) Inadequate stirring of paint or primer.
(3) Coating at incorrect time intervals.
(4) Application under adverse conditions.
(5) Bad application.

B. Spray Dust - Spray dust caused by atomized particles drying before reaching surface being painted fail to flow as a continuous film. Usual causes are incorrect air pressure or distance gun is held from work.

C. Sags and Runs - Excess paint causes wet paint film to move by gravity and presents a sagging appearance. Incorrect viscosity, air pressure, and gun handling, or inadequate surface preparation are frequent causes.

D. Spray Mottle - Sometimes known as orange peel or pebble, is caused by incorrect paint viscosity, air pressure, spray gun setting, or the distance the gun is held from work.

E. Blushing is one of the most common troubles. It appears as clouding or blooming of paint film. It is more common with cellulose than synthetic materials. It may be caused by moisture in air supply line, adverse humidity, drafts, or sudden temperature changes.

9. Storage

A. Store paint, enamel, and other finishing material in dry storage away from direct sunlight and heat. Mark each container with a code for identification.

B. Storage facilities must comply to Occupational Safety and Health Act (OSHA) requirements regarding air circulation, lighting, and fire protection. Lock storage facilities to prevent children and unauthorized personnel entry.

C. Invert pigmented materials every inventory so pigments will not pack to can bottom. Properly dispose of empty containers.

D. Use older materials first. Useful life of some finishes is limited.

E. Storage area temperatures must be approximately 50-90°F. If finishes are stored in temperature extremes, allow them to return to room temperature before using.

10. Painting Facility

WARNING: DO NOT BREATHE PAINT FUMES. FUMES DEPLETE THE OXYGEN SUPPLY REQUIRED BY THE BODY.

A. Painting facilities must conform to local, state, and OSHA standards with respect to air circulation, exhaust emissions, lighting, and fire protection.

B. Provide sufficient air movement in painting area so there is only a slight finishing material odor. Exhaust fans must be belt-driven and located near floor level. Locate fan's motor away from fumes.

C. All spraying area personnel must wear approved respiration safety equipment.

11. Aircraft Finish Care

See Aircraft Finish Care, 12-00-00.

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CHAPTER

21

ENVIRONMENTAL SYSTEMS

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CHAPTER 21

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
21-List of Effective Pages	1	Nov 30/17
	2	Nov 30/17
21-Table of Contents	1	Nov 30/17
	2	Nov 30/17
	3	Nov 30/17
	4	Nov 30/17
	5	Nov 30/17
	6	Nov 30/17
21-00-00	1	Nov 30/17
	2	Dec 15/14
	3	Dec 15/14
	4	Dec 15/14
	5	Dec 15/14
	6	Dec 15/14
21-20-00	1	May 31/11
	2	Sep 15/09
	3	May 31/11
	4	Dec 15/14
	5	Dec 15/14
	6	Dec 15/14
	7	Dec 15/14
	8	Dec 15/14
	9	Dec 15/14
	10	Nov 30/17
	11	Nov 30/17
	12	Nov 30/17
	13	Nov 30/17
	14	Nov 30/17
	15	Nov 30/17
	16	Nov 30/17
	17	Nov 30/17
	18	Nov 30/17
	19	Nov 30/17
	20	Nov 30/17
21-30-00	1	Dec 15/14
	2	Dec 15/14
	3	Dec 15/14
	4	Dec 15/14
	5	Dec 15/14
	6	Dec 15/14
	7	Dec 15/14
	8	Dec 15/14
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	25	Nov 30/17
	26	Nov 30/17
	27	Nov 30/17
	28	Nov 30/17
	29	Nov 30/17
	30	Nov 30/17
21-40-00	1	Nov 30/17
	2	Feb 15/14
	3	Nov 30/17
	4	May 31/11
	5	May 31/11
	6	May 31/11
	7	May 31/11
	8	May 31/11

CHAPTER 21

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
21-50-00	1	Nov 30/17			
	2	Jul 1/10			
	3	Sep 15/09			
	4	Jul 1/10			
	5	Jul 1/10			
	6	Nov 30/17			
	7	Nov 30/17			
	8	Sep 15/09			
	9	Nov 30/17			
	10	Sep 15/09			
	11	Nov 30/17			
	12	Jul 1/10			
	13	Jul 1/10			
	14	May 31/11			
	15	May 31/11			
	16	Nov 30/17			
	17	Nov 30/17			
	18	Sep 15/09			
	19	Nov 30/17			
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	34	Nov 30/17			
	35	Dec 15/14			
	36	Nov 30/17			
	37	Nov 30/17			
	38	Dec 15/14			
	39	Dec 15/14			
	40	Nov 30/17			
21-60-00	1	Dec 15/14			
	2	Sep 15/09			

CHAPTER 21 - ENVIRONMENTAL SYSTEMS

TABLE OF CONTENTS

CHAPTER 21 - ENVIRONMENTAL SYSTEMS

TABLE OF CONTENTS (continued)

CHAPTER 21 - ENVIRONMENTAL SYSTEMS

TABLE OF CONTENTS (continued)

CHAPTER 21 - ENVIRONMENTAL SYSTEMS

TABLE OF CONTENTS (continued)

CHAPTER 21 - ENVIRONMENTAL SYSTEMS

TABLE OF CONTENTS (continued)

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GENERAL

1. PA-46-350P Aircraft

The environmental system consists of: a compressor bleed air and conditioning system, the ventilating air system, a supplemental electric cabin heater, an air conditioning system, the cabin air distribution system, and the pressurization and control system.

Switches (and push-pull knobs in S/N's 4636001– 4636374) to control and regulate the various systems, except the pressurization system, are located:

- below the control wheel on the lower left and lower center sections of the copilot's instrument panel;
- on the right switch panel above the right radio stack in S/N's 4636021 thru 4636374;
- on the switch panel to the right of the MFD, in Avidyne Entegra and Garmin Integrated Avionics System (IAS) G1000/G1000 NXi equipped airplanes.

Compressor bleed air from the engine turbochargers supplies air for heating the cabin during flight and ground operations and for pressurization. The bleed air is first routed through an air-to-air heat exchanger, and then into the cabin through the lower left and right cabin side panel ducts. The heat exchanger utilizes ambient ram air to cool the bleed air, or hot air from an exhaust shroud to heat the bleed air. Desired cabin comfort is maintained by using the CABIN TEMP push-pull knob to manually adjust a flapper-type control valve located forward of the firewall. The position of this valve will allow ambient air, or hot air, or a mixture of both, to enter the heat exchanger.

The cabin pressurization system outflow valve(s) provides the means by which smoke and impurities are vented from the cabin.

Cabin ventilating air during ground or unpressurized low altitude flight operations is provided by the ambient ram air source to the bleed air heat exchanger. An electric vane-axial ventilation/defog blower, located in the left cabin air inlet duct below the forward baggage compartment floor, is used to produce an air flow to the windshield defogger, and to supplement the inflow of ventilating air during ground operations. The blower is activated by selecting the VENT DE-FOG switch ON. Incoming ventilating air can be heated by mixing it with hot air from the exhaust shroud.

NOTE: If electric supplemental heat is not used, maximum cabin heat for ground operations and unpressurized low altitude flight will be obtained with the CABIN PRESS control knob full out.

The supplemental electric heater consists of a resistance type heat element, a dual hermetically sealed bimetallic type overtemperature protection, a power relay, and a 35 amp in line current limiter fuse. Its function is to provide additional heat for maintaining desired cabin comfort during ground or flight operations under temperature conditions when fully heated bleed air or ventilating air is inadequate. When an external power source is used, the supplemental heater can also be used to preheat the cabin prior to engine start. See Section 2 of the Pilot's Operating Handbook (POH) for limitations on use of the supplemental heater.

The supplemental heater heat element is installed forward of the pressure bulkhead in the left bleed air duct immediately downstream of the ventilation/defog blower. Because the ventilation/defog blower must be operating whenever supplemental heat is used, both the VENT DE-FOG and AUXILIARY CABIN HT switches must be ON to supply power to the heating element.

Both the heater control circuit and the vent/defog fan circuit utilize the 10 amp VENT DE-FOG circuit breaker located on the pilot's aft circuit breaker panel. Heater element power is supplied from the battery master solenoid through the 35 amp heater fuse and the heater power relay. The 35 amp heater fuse is not accessible to the pilot. The electrical load imposed by the heater and the vent/defog fan is approximately 40 amps. Operation is limited to airplanes with both alternators functioning.

Cabin air conditioning is provided by a vapor cycle system. The compressor is belt driven by the engine. Condenser cooling airflow is provided by a continuous duty motor driven fan. Cabin air is recirculated across the evaporators to provide cool air at each seat outlet.

The condenser and its cooling air fan are located in the tailcone immediately aft of the rear pressure bulkhead. Cooling air from outside the tailcone is drawn into the cooling air duct through a flush opening in the skin, routed across the condenser coil, and discharged overboard through the tailcone exit opening.

Two recirculation blowers and evaporator assemblies are located aft of each rear seat below the rear baggage compartment floor. The recirculation blowers draw air into each evaporator coil through grills in the floor structure behind the rear seats and discharges it into the upper left and right cabin side panel ducts. Adjustable eyeball outlets are located at each seat in the airplane.

The AIR COND and BLOWER HIGH & LOW switches, located as part of the environmental switch panel, are used to control the air conditioning system.

In S/N's 4636001 thru 4636131, when the AIR COND/OFF/BLWR switch is positioned to AIR COND, the compressor belt drive is electrically clutched, the condenser blower motor relay is closed, and both recirculation blowers are activated. The recirculation blowers can be operated independently of the air conditioner by setting the AIR COND/OFF/BLWR switch to BLWR. In either situation, the REC BLWR switch is used only to select a HI or LO recirculation blower motor speed. The AIR COND/OFF/BLWR switch must be set to the center OFF position to shut off the recirculation blower motors.

In S/N's 4636132 and up, when the AIR COND switch is selected ON, the compressor belt drive is electrically clutched, the condenser blower motor relay is closed, and both recirculation blowers are activated. The recirculation blowers can be operated independently of the air conditioner by selecting the BLOWER HIGH or LOW ON. In either situation, the BLOWER switches are used only to select a HIGH or LOW recirculation blower motor speed. When selecting between BLOWER HIGH and BLOWER LOW the switch currently "ON" should be deselected to "OFF" before selecting the other "ON".

Overcurrent protection is provided by the 10 amp CABIN BLOWERS (early airplanes) or 15 amp CABIN FANS (later airplanes), 5 amp AIR CONDITIONER CONTROL, and 25 amp AIR CONDITIONER POWER circuit breakers in the nonessential bus section of the pilot's forward circuit breaker panel.

The refrigerant portion of the system incorporates a receiver dryer, a sight gauge, suction and discharge service valves, and 265 psi high pressure and 40 psi low pressure switches. Should the compressor discharge pressure increases above 265 psi, or decrease below 40 psi, the applicable pressure switch will open, disengaging the compressor clutch.

Air for cabin pressure is obtained from the engine turbocharger induction air system through two sonic venturi tubes. Bleed air is routed through the bleed air heat exchanger for the temperature conditioning to provide the desired cabin comfort level. Ram ambient air is routed across the heat exchanger to cool the bleed air, and hot ambient air from the heat muff is routed across the heat exchanger to heat the bleed air. Mixtures of ram ambient and heated ambient air may also be selected.

Cabin air is controlled by a push-pull knob labeled CABIN PRESS located beneath the control wheel on the pilot's instrument panel. Bleed air for pressurizing the cabin is provided when the control is fully in. Unpressurized ambient air is provided for ventilating the cabin when the control is fully out. This control operates three valves: the bleed air shutoff valve, the bleed air dump valve, and the ram air selector valve. When pushed fully in, the bleed air shutoff valve is open, the bleed air dump valve is closed, and the ram air selector valve is positioned to route ambient air across the bleed air heat exchanger. When the control is pulled completely out, the bleed air shutoff valve is closed, the bleed air dump valve is open, and the ram air selector valve is positioned to route ambient air into the conditioned air ducts through the check valve and into the cabin.

A. Pneumatic Presurization Control System

S/N's 4636001–4636651 less 4636633

The cabin pressurization and control system consists of an isobaric outflow valve, a safety outflow valve, cabin altitude and rate selector, electronically operated vacuum solenoid valve, surge tank, and associated interconnecting plumbing and wiring. Cabin altitude, differential pressure, and rate of change are displayed on a single three inch diameter indicator, or on the MFD in airplanes equipped with Garmin G1000. Should cabin pressure altitude exceed 10,000 feet:

- the CABIN ALTITUDE annunciator will illuminate to warn the pilot;
- except in Garmin G1000 equipped airplanes, where the red CABIN ALT 10000 CAS message will illuminate in the MFD to warn the pilot.

Controls and switches needed to operate the cabin pressurization system are located on the lower section of the pilot's instrument panel and on the lower copilot's instrument panel. In addition to the CABIN PRESS and CABIN TEMP controls, are the cabin pressure and rate controller located:

- in S/N's 4636001–4636020, just above the tie bus circuit breakers, and the CABIN PRESS DUMP/NORM switch located on the pilot's main switch panel.
- in S/N's 4636021–4636374, just above the fuel selector control, and the CABIN PRESS DUMP/NORM switch located on the right overhead switch panel.
- in airplanes equipped with Avidyne Entegra, to the left of the standby altimeter, and the CABIN PRESS DUMP/NORM switch located on the right overhead switch panel.
- in airplanes equipped with Garmin G1000, below the standby airspeed indicator, and the CABIN PRESS DUMP/NORM switch located on the right overhead switch panel.

For pressurized flight, set the cabin pressure controller at 500 feet above the airport pressure altitude, CABIN PRESS control knob full in and the CABIN PRESS DUMP/NORM switch to NORM. The rate of cabin ascent and descent change is controlled with the rate knob (left lower corner of the cabin pressure controller), and may be adjusted between approximately 200 and 2000 feet per minute, as desired. Setting the rate knob arrow to the 9 o'clock position provides a cabin altitude rate of change of approximately 500 feet per minute. This position gives a comfortable rate for normal operations.

In S/N's 4636001–4636459, 4636461–4636462, and 4636481:

Below/next to the cabin pressure controller (or pilot's PFD, if Avidyen Entegra equipped), a triple indicator simplifies monitoring the system's operation. The triple indicator displays the cabin altitude, cabin rate of change and the differential pressure between the cabin and the outside atmosphere. Maximum cabin differential pressure is 5.5 psi.

A CABIN ALTITUDE warning light on the annunciator display warns the pilot when the cabin altitude is above 10,000 feet. Cabin pressure is automatically regulated to a maximum of 5.5 psi pressure differential. Should the cabin outflow valve malfunction, the cabin safety valve will maintain a maximum of 5.6 cabin differential pressure.

In airplanes equipped with Garmin G1000 :

A CABIN ALT 10000 warning CAS message is illuminated when the cabin altitude is above 10,000 feet. Cabin pressure is automatically regulated to a maximum of 5.6 psi pressure differential. Should the cabin outflow valve malfunction, the cabin safety valve will maintain a maximum of 5.6 psi cabin differential pressure. Should the cabin differential pressure remain above 5.6 psid for greater than 30 seconds, or if it reaches 5.8 psid, a Master Warning will be triggered, with a red DIFF PSI indication (on the MFD) and a repeating aural chime.

The landing gear squat switch, on the left main landing gear, prevents the cabin from being pressurized while the airplane is on the ground.

The CABIN PRESS DUMP/NORM switch, when set to DUMP, electrically opens a solenoid valve allowing vacuum suction pressure to open the safety valve and rapidly dump cabin pressure to ambient pressure.

For unpressurized flight the CABIN PRESS control should be pulled fully out. Setting the CABIN PRESS/DUMP/NORM switch to DUMP will provide maximum airflow through the cabin. Cabin temperature will continue to be controlled by the CABIN TEMP control.

B. Electro-Mechanical Presurization Control System

S/N's 4636633, 4636652 and up.

The cabin pressurization and control system is comprised of two identical outflow valves, a controller, and a cabin pressure transducer and associated interconnecting plumbing and wiring. The controller is mounted on a shelf in the rear of the airplane near the aft pressure bulkhead. Both outflow valves are mounted directly on the aft pressure bulkhead facing into the cabin. The cabin pressure transducer is mounted in the cockpit near the instrument panel.

Cabin pressure is controlled by two electromechanical outflow valves located on the aft pressure bulkhead. These valves provide identical functions of maintaining a desired pressurization schedule during all phases of flight without exceeding the maximum differential pressure of 5.5 +/-0.1 psi. If the cabin pressure control system develops a communications fault during flight:

- A CPCS FAULT caution CAS message is posted.
- The landing field elevation and weight on wheels sensors are lost, causing the cabin to remain pressurized after landing. Accordingly, the pilot must verify that the cabin is fully depressurized prior to landing.

If the cabin pressure control system develops a communications failure on the ground, a CPCS FAIL caution CAS message is posted. The outflow valves will remain open and the cabin will not pressurize. If a CPCS FAIL caution CAS message is experienced during flight, the outflow valves will close, allowing the system to continue to operate in a degraded mode.

The cabin pressurization system controls, switches, and displays are as follows:

- CABIN PRESS push-pull knob;
- CAB PRES DUMP / NORM switch;
- Cabin Altitude, Differential Pressure, and Rate of Climb EIS indications on MFD; and
- Destination Field Elevation (DEST ELV) in TMR/REF window on PFD.

The only action required by the pilot during normal operation is to input the destination airport elevation by selecting the TMR/REF softkey on either PFD and entering the destination airport elevation in the DEST ELV field. If a new destination elevation is not entered, the last value entered will be used, which could result in the airplane landing while still pressurized.

The controller automatically schedules cabin pressure from takeoff to landing. If the pilot changes his destination in-flight, the landing field elevation information must be manually updated in the PFD. If communication fails in flight, the controller will use the last field elevation that was output. In flight, the cabin altitude and rate are automatically scheduled to minimize the pressure changes experienced by the passengers and crew. During descent, the controller automatically schedules a comfortable cabin altitude rate to allow the cabin to reach landing field pressure elevation upon landing. After landing, the controller ensures that the cabin remains safely depressurized.

Cabin altitude, rate, and differential information are sensed by the cabin pressure transducer and are presented to the pilot on the avionics displays.

A CABIN ALT 10K warning CAS message is illuminated when the cabin altitude is above 10,000 feet. Cabin pressure is automatically regulated to a maximum of 5.6 psi pressure differential. Should the cabin differential pressure remain above 5.6 psid for greater than 30 seconds, or if it reaches 5.8 psid, a Master Warning will be triggered, with a red DIFF PSI indication (on the MFD) and a repeating aural chime.

When the CABIN PRESSURE DUMP switch is set to DUMP, the controller sets both outflow valves to full open and the valves remain open until the switch is disengaged or the maximum cabin altitude is reached.

For unpressurized flight the CABIN PRESS control should be pulled fully out. Setting the CABIN PRESS/DUMP/NORM switch to DUMP will provide maximum airflow through the cabin. Cabin temperature will continue to be controlled by the CABIN TEMP control.

2. PA-46R-350T Aircraft

The environmental system consists of a cabin air distribution system, ventilating air system and air conditioning system.

Environmental control is available in the two following configurations.

A. Ram Air Heat and Defrost

See 21-40-00.

Fresh air from a heater inlet duct on the lower right cowling passes through a shroud around the muffler and is ducted to the cabin.

On S/N's 4636001 and up, and 4692001 and up, the desired cabin comfort is maintained by using the CABIN TEMP push-pull knob to manually adjust a flapper-type control valve located forward of the firewall.

CAUTION: TO AVOID OVERHEATING OF THE EXHAUST HEAT SHROUD, ALWAYS OPEN THE CABIN VENT CONTROL KNOB (IF EQUIPPED) BEFORE OPENING THE CABIN TEMP CONTROL KNOB.

On S/N's 4692165 and up, with G1000, cabin ventilation is controlled by the CABIN VENT control knob. Pulling the CABIN VENT control knob out from the panel opens a valve and allows fresh air to flow into the cabin. With the CABIN VENT control open, desired cabin comfort is maintained by using the CABIN TEMP push-pull knob to manually adjust a flapper-type control valve located forward of the firewall.

The position of this temperature control valve will allow ambient air, or hot air, or a mixture of both to enter the cabin air distribution system, in addition to the windshield defroster when selected by pulling the DEFROST knob. Control knobs are located to the right of the flap control.

B. Air Conditioning

See 21-50-00.

The air conditioning system uses a two-speed blower to recirculate cabin air through an evaporator and filter located in the aft fuselage. The cooled air is then ducted back into the cabin through six eyeball vents.

A compressor mounted on the front of the engine takes heat-laden, vaporized, refrigerant from the evaporator and pumps it to a condenser mounted in the aft fuselage where it is cooled to a liquid state and pumped back into the evaporator.

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DISTRIBUTION

The cabin air distribution system consists of left and right side panel ducting, windshield defrost, foot warmer (if installed) and ventilation blowers. The side panel ducts provide for overall air distribution throughout the length of the cabin as well as individual controllable air outlets at each seat (eyeball outlets). The defrost control will allow part of the bleed/heated air to be diverted to the windshield defrost outlet. Separate and individual controls for the two foot warmer outlets (if installed) are also provided. The outlets are located between the rudder pedals on each side of the cockpit. The ventilation blowers supply airflow to the portion of the sidewall ducts containing the individual seat outlets (eyeballs).

On PA-46-350P aircraft, cabin ventilating air, for ground and unpressurized flight operation, is supplied from the ambient air source to the bleed air heat exchanger through a ram air selector valve and check valve. This will supplement air flow primarily in ground operation. This air source is capable of being heated by mixing with hot air from the exhaust shroud.

On PA-46R-350T aircraft, cabin ventilating air, for ground operation, is supplied from the ambient air source which is capable of being heated by mixing with hot air from the exhaust shroud.

NOTE: See 21-00-00 for environmental control description.

1. Ground Blower

   A vane-axial ground blower is installed in the left duct below the forward baggage compartment floor. This blower supplements air flow during ground operation. Flapper-type check valves (PA-46-350P only) are located at the forward pressure bulkhead.

   A. Removal

      (1) Loosen the two clamps that secure the flexible adapter ducts to the blower.
      (2) Loosen the two clamps that secure the flexible adapter ducts to the inlet and outlet ducts.
      (3) Slide the aft flexible adapter duct rearward over the outlet duct.
      (4) Disconnect the electrical leads from the blower.
      (5) Slide the blower aft and down to remove it.

   B. Installation

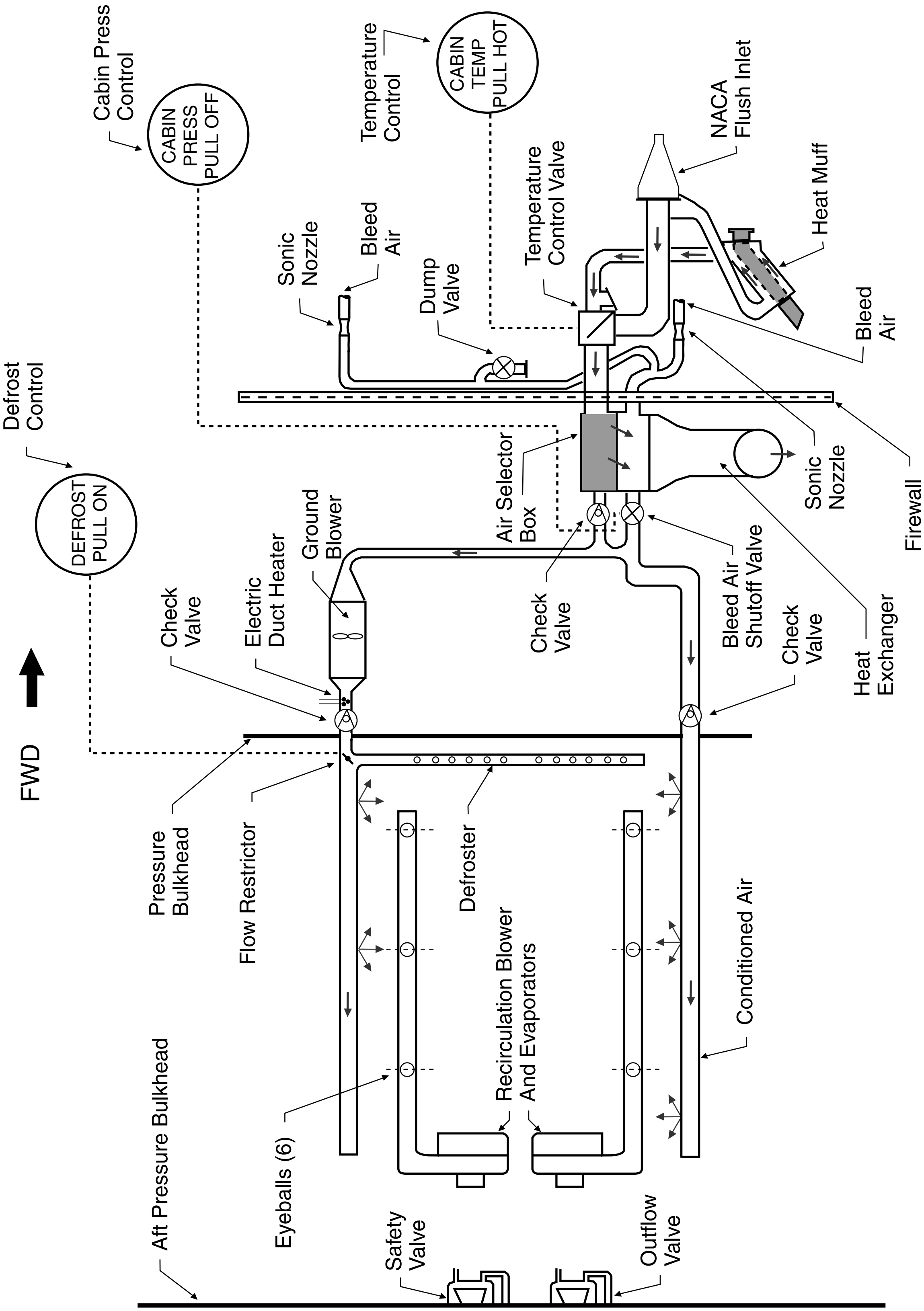
      (1) Position blower in place, ensuring that the airflow arrow is pointing aft.
      (2) Slide the aft flexible adapter duct forward onto blower and secure it with the clamp.
      (3) Secure the forward flexible adapter duct to the blower with the clamp.
      (4) Tighten clamps that secure the flexible adapter ducts to the inlet outlet ducts.
      (5) Connect leads from the blower.

2. Cabin Recirculation Blowers

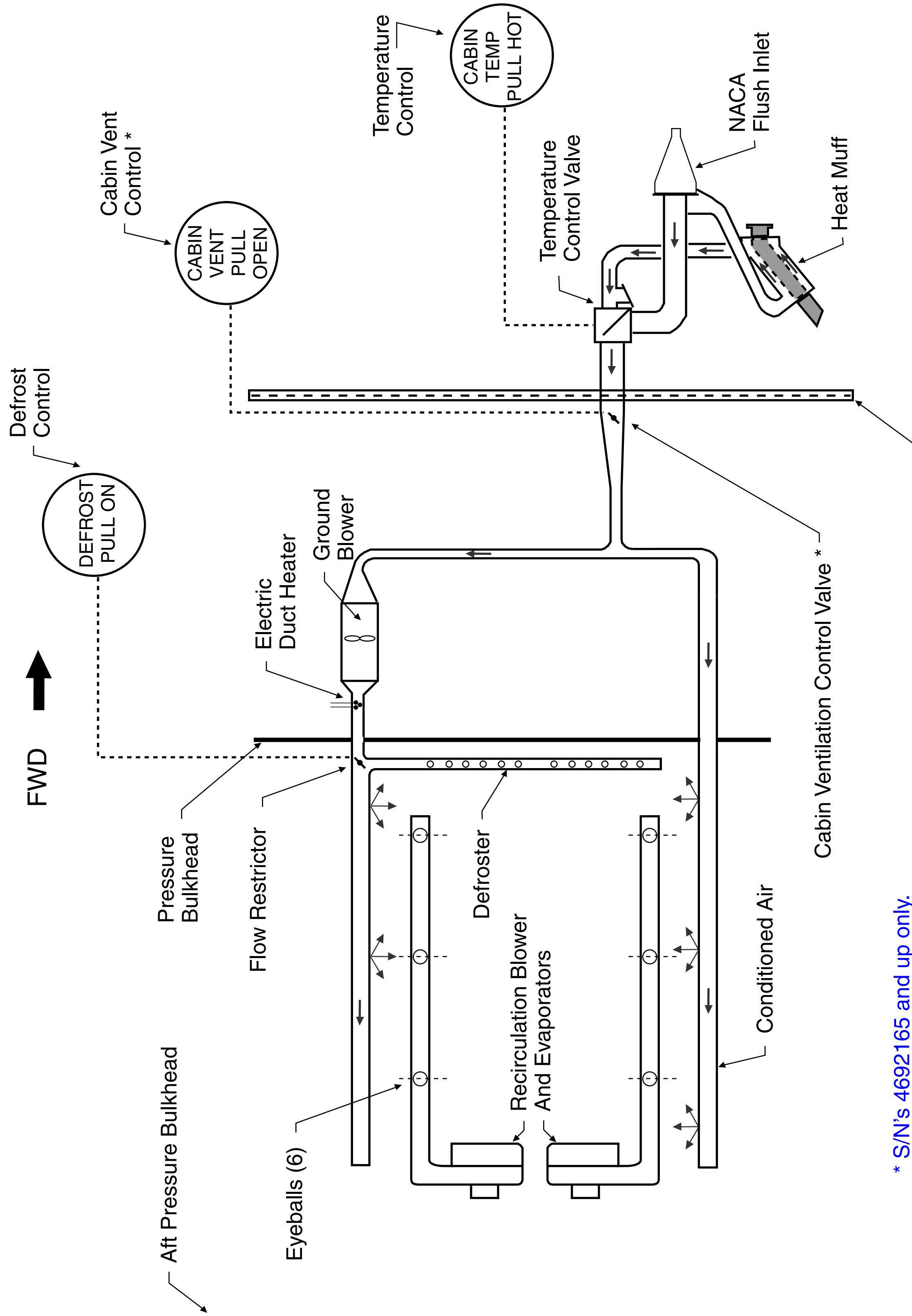
   The left and right cabin recirculation blowers are beneath the rear baggage compartment floor between F.S. 233.87 and F.S. 241.50.

   A. Removal

      (1) Remove the rear seats, the rear baggage compartment carpet, and the two large access panels in the rear baggage compartment floor.
      (2) Disconnect the duct from the outboard side of the blower.
      (3) Disconnect the electrical lead going to the blower motor.
      (4) Remove the bolts, nuts and washers that attach the blower to the evaporator shell.
      (5) Remove the blower from its cavity.



Air Distribution System
Figure 1 (Sheet 1 of 2)



Air Distribution System
Figure 1 (Sheet 2 of 2)

Effectivity
4692001 and Up

B. Installation

(1) Position the blower in place on the evaporator shell and secure it with the bolts, nuts and washers.

(2) Connect the electrical leads going to the blower motor. (See Chapter 91 for electrical schematic.)

(3) Connect the duct to the outboard side of the blower.

(4) Reinstall access panels, carpeting, and rear seats.

3. Bleed Air Check Valve (PA-46-350P Only)

See “Figure 1”.

A. Removal

(1) On left hand side, remove the six bolts attaching cabin air supply tube and auxiliary electric heater to forward pressure bulkhead.

(2) Break seal and remove check valve.

(3) On right hand side, loosen band clamps on flexible hoses connecting bleed air tubes.

(4) Remove the six bolts connecting cabin air supply tube and bleed air supply tube to forward pressure bulkhead.

(5) Break seal and remove check valve.

B. Installation

(1) Apply GE RTV 1508 sealant to mating surface of check valves.

(2) On left hand side, position valve and gaskets (with flapper on aft side) between auxiliary heater and forward pressure bulkhead.

(3) Secure valve, auxiliary heater, and cabin air supply tube to forward pressure bulkhead with six bolts.

(4) On right hand side, position valve and gaskets (with flapper on aft side) between bleed air supply tube and forward pressure bulkhead.

(5) Secure valve, bleed air supply tube, and cabin air supply tube to pressure bulkhead with six bolts.

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4. Avionics Cooling

See “Figure 2” and “Figure 3”.

NOTE: Actual equipment served will vary depending on optional equipment installed.

A. 1995–2010

See “Figure 2”

(1) Avionics Cooling Blower

The avionics cooling blower is installed on the right side of the fuselage, behind the instrument panel, at FS 108.00. The cooling air is directed through air duct hose to installed avionics equipment.

(a) Removal

1) Disconnect duct hose from blower ports.

NOTE: Make note/tag each duct hose to enable reinstallation to the correct port.

2) Disconnect blower motor wires at connector.

3) Remove screws, washers and nuts that attach the blower assembly to its mounting bracket.

4) Remove blower.

(b) Installation

1) Attach the blower assembly to the mounting bracket with the screws, washers and nuts.

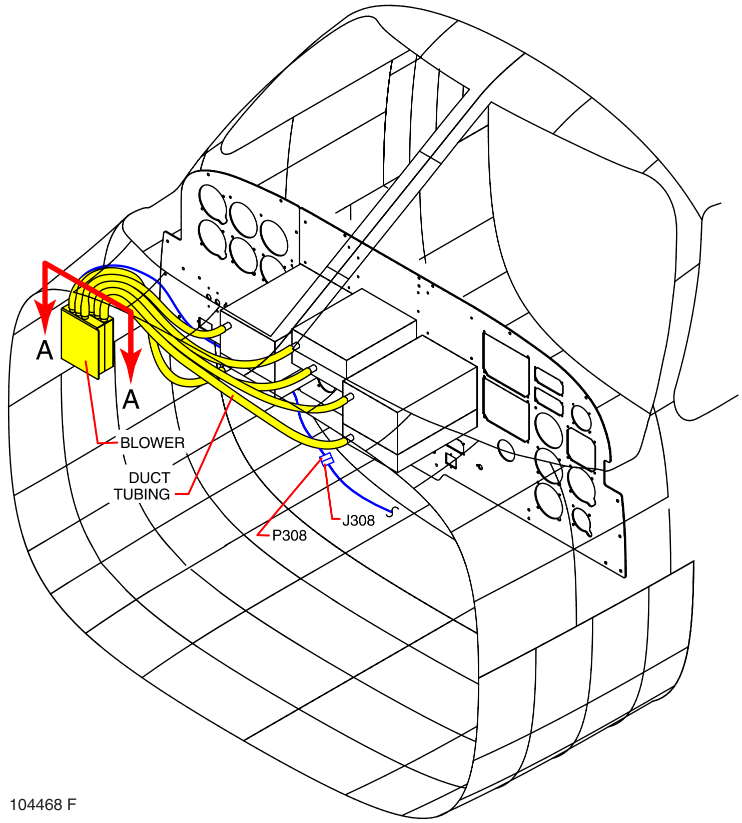
2) Reconnect the blower motor wires.

3) Attach duct hoses to blower ports. Ensure that the duct ends overlap the ports on top of the blower one-half inch minimum. Secure with Ty-Raps.

(2) Muffin Fan

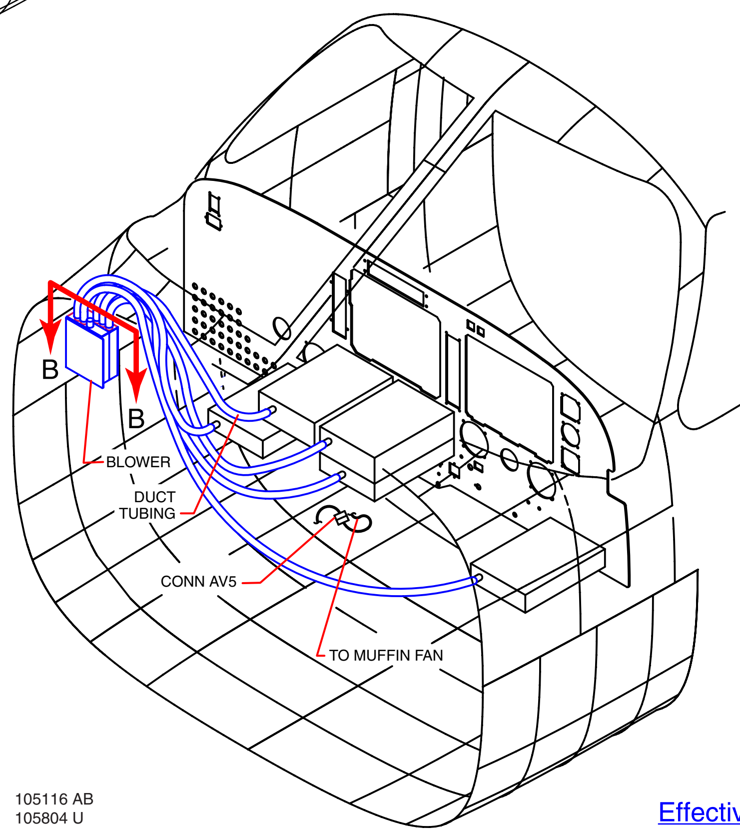
See “Figure 2”.

In S/N’s 4636299 and 4636314–4636459, 4636461–4636462, and 4636481; and in S/N’s 4692001–4692133, 4692141, 4692149 and 4692153; a muffin fan is installed in the control pedestal.



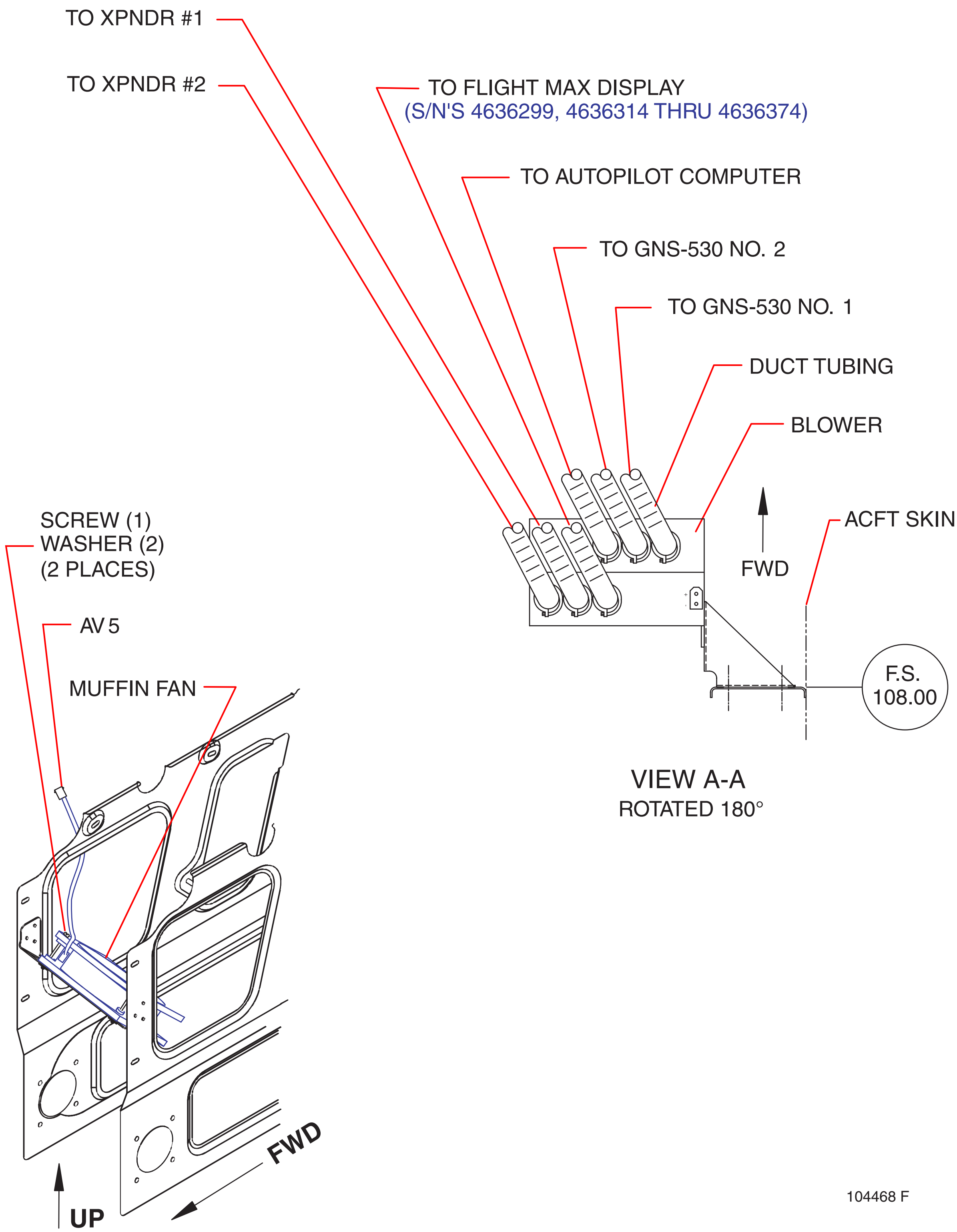
PA-46-350P
SHOWN WITH CONVENTIONAL GAUGES

PA-46R-350T
SHOWN WITH AVIDYNE ENTEGRA



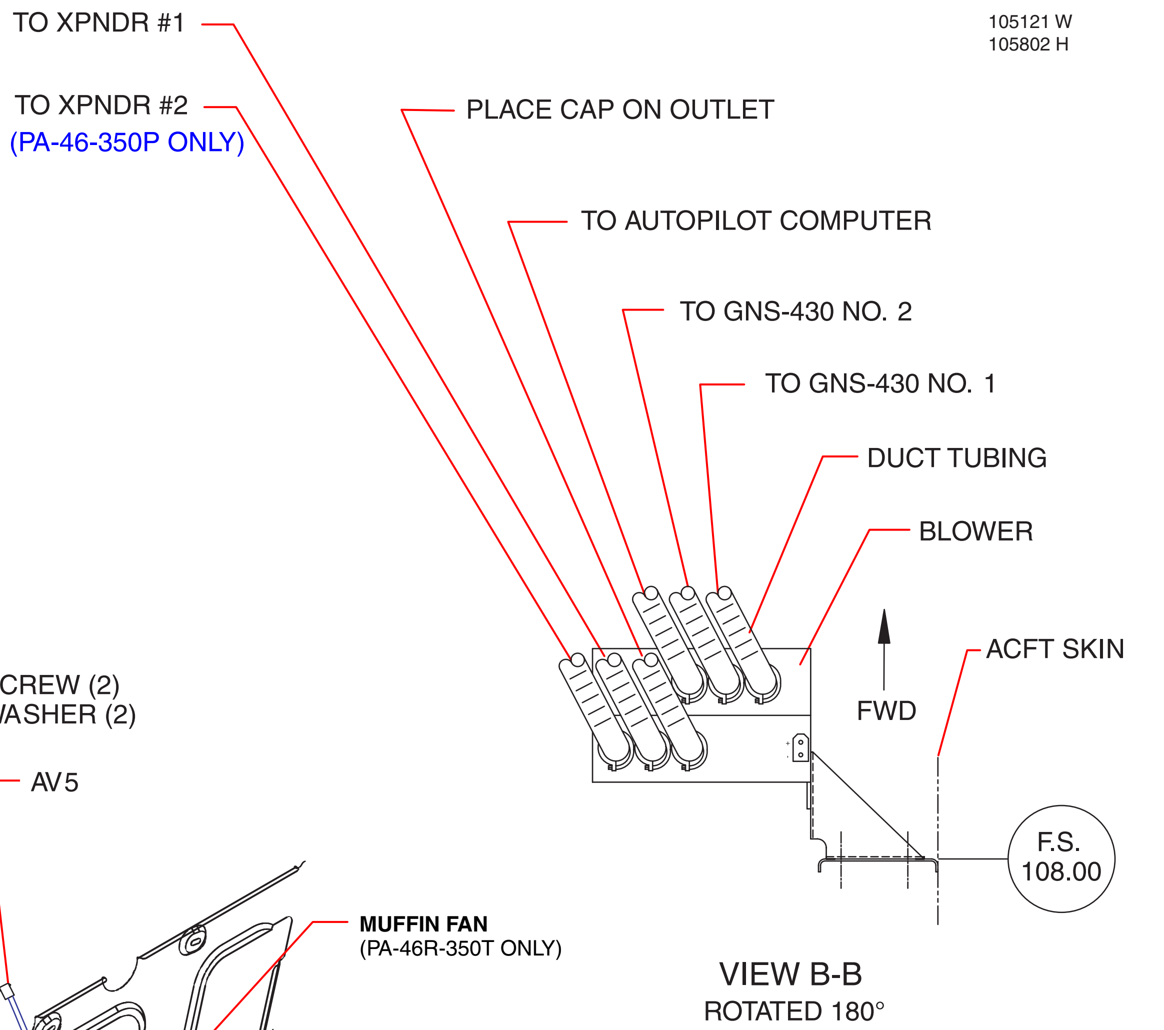
Effectivity
4636001–4636459
4636461–4636462, 4636481
4692001–4692133, 4692141
4692149 and 4692153

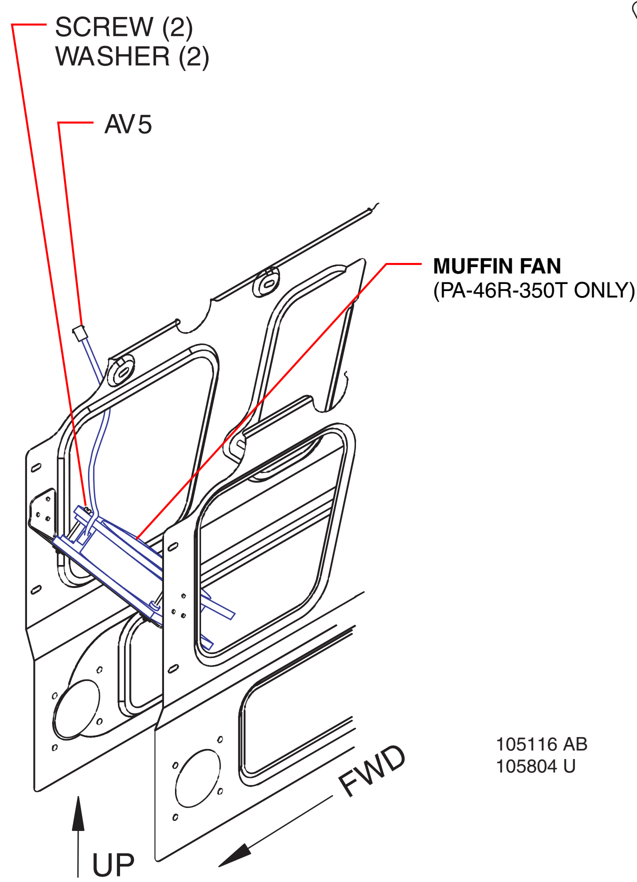
Avionics Cooling (1995–2010)
Figure 2 (Sheet 1 of 3)



Effectivity
4636299,
4636314 thru 4636374

Avionics Cooling (1995–2010)
Figure 2 (Sheet 2 of 3)





Avionics Cooling (1995–2010)
Figure 2 (Sheet 3 of 3)

Effectivity
4636375–4636459
4636461–4636462, 4636481
4692001–4692133, 4692141
4692149 and 4692153

B. 2010 and up

See “Figure 3” on page 212011.

(1) S/N’s 4636460, 4636463–4636651 less 4636481 and 4636633; and S/N’s 4692134 and up less 4692141, 4692149, and 4692153

Four (4) cooling fans are installed behind and under the instrument panel. Two (2) fans are rack mounted with one (1) behind each PFD. The third (3rd) is mounted to an adapter plate attached to the side of the equipment shelf behind the MFD. The fourth (4th) fan is remotely mounted under the equipment rack behind the MFD. This fan uses air duct hoses to supply air to the GTX 33’s and the GIA 63W’s.

(a) Rack-Mounted Fans (Safe 328) (2 ea.)

1) Removal

a) Remove the respective PFD. (See Components in 34-25-01.)

b) Disconnect wire connector.

c) Remove six (6) screws, three (3 ea.) top and bottom.

d) Slide fan and rack mount out of rack and remove three (3) screws that attach fan to the mount.

2) Installation

a) Reattach fan to rack mount with three (3) screws.

b) Slide fan and rack mount into the rack and secure with six (6) screws, three (3 ea.) top and bottom.

c) Reconnect wire connector.

d) Reinstall PFD.

(b) MFD Fan (Safe 328)

1) Removal

a) Remove the MFD. (See Components in 34-25-01.)

b) Disconnect wire connector.

c) Remove three nuts and washers securing fan to adapter plate.

d) Remove fan.

2) Installation

a) Slide fan over studs on adapter plate and secure with three washers and nuts.

b) Reconnect wire connector.

c) Reinstall MFD.

(c) Remote Fan (Safe 528)

1) Removal

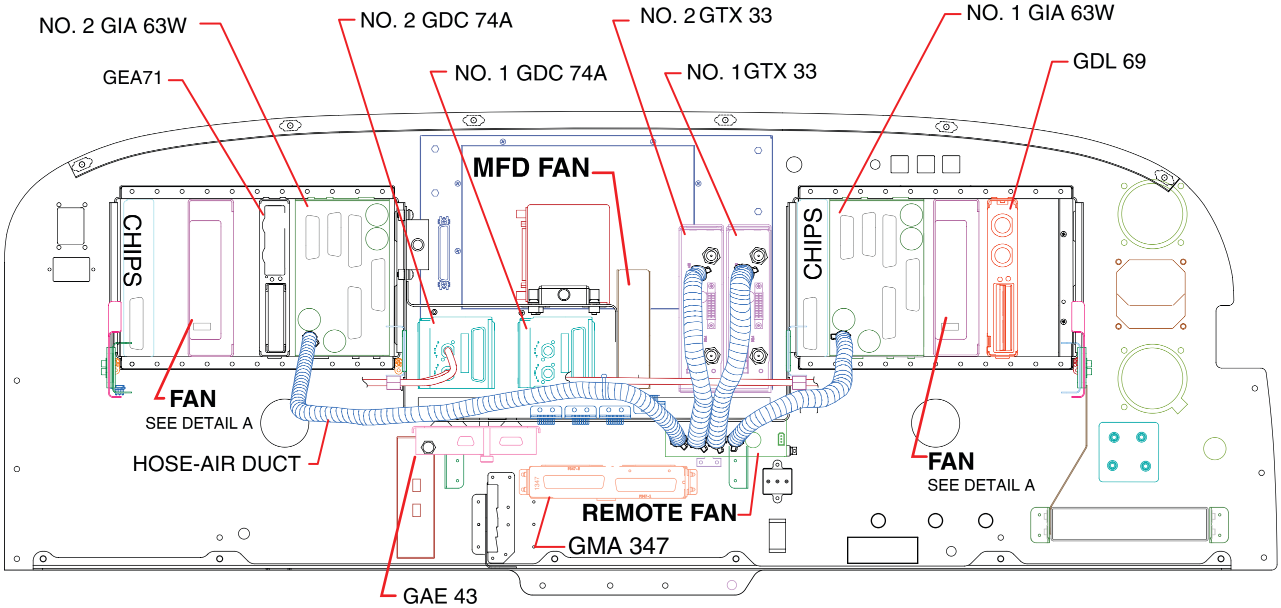
a) Disconnect air duct hoses and wire connector.

NOTE: Make note/tag each duct hose to enable reinstallation to the correct port.

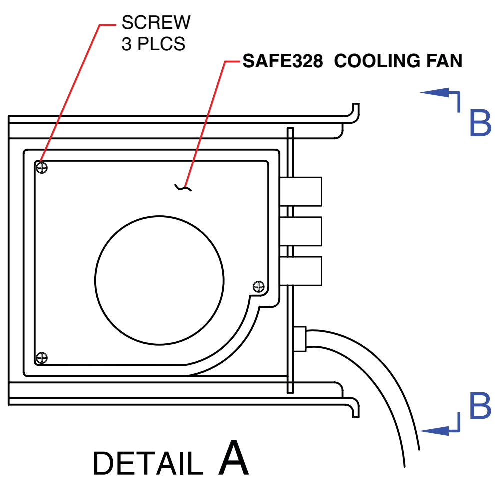
b) Remove four screws securing fan to bottom of the equipment rack.

c) Remove fan.

105550

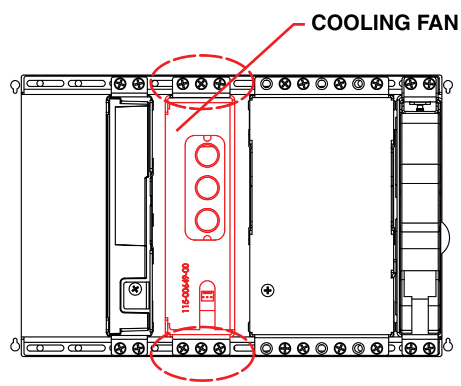


AIR HOSE DUCTING
LOOKING AFT



DETAIL A

LOOKING RIGHT



VIEW B - B

2 PFD COOLING FANS
SYSTEM RACK LAYOUT TYPICAL
LOOKING FWD

Avionics Cooling (2010 - 2014)
Figure 3

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153

2) Installation

a) Position fan to underside of equiment rack and secure with four screws.

b) Reconnect wire connector and air duct hoses (see “Figure 3”, Sheet 1).

NOTE: Cap any unused fan ports.

c) Reinstall MFD.

(2) S/N’s 4636633, 4636652 - 4636715, 4636717 - 4636719

See ”Figure 4” on page 212014.

Six (6) cooling fans may be installed. Five (5) cooling fans are installed behind or under the instrument panel. One (1) cooling fan may be installed on the aft equipment shelf.

(a) Instrument Panel Cooling Fans

Two (2) fans (Safe 328) are rack mounted with one (1) behind each PFD. A third (3rd) (Safe 328) is mounted to an adapter plate on the shelf assembly behind the MFD. These three fans blow directly on the equipment cooled.

The fourth (4th) fan (Safe 328) is mounted to a bracket under the equipment rack below the #2 GIA 63W. This fan uses air duct hoses to supply air to the GMA 350 and Aspen Avionics EBD.

1) Rack-Mounted Fans (Safe 328) (2 ea.)

a) Removal

1] Remove the respective PFD. (See Components under in 34-25-01.

2] Disconnect wire connector.

3] Remove six (6) screws, three (3 ea.) top and bottom.

4] Slide fan and rack mount out of rack and remove three (3) screws that attach fan to the mount.

b) Installation

1] Reattach fan to rack mount with three (3) screws.

2] Slide fan and rack mount into the rack and secure with six (6) screws, three (3 ea.) top and bottom.

3] Reconnect wire connector.

4] Reinstall PFD.

2) MFD Fan (Safe 328)

a) Removal

1] Remove the MFD. (See Components in 34-25-01.

2] Disconnect wire connector.

3] Remove three (3) nuts and washers securing fan to adapter plate.

4] Remove fan.

b) Installation

1] Slide fan over studs on adapter plate and secure with three (3) washers and nuts.

2] Reconnect wire connector.

3] Reinstall MFD.

3) Remote Fan (Safe 528)

a) Removal

1] Disconnect air duct hoses and wire connector.

NOTE: Make note/tag each duct hose to enable reinstallation to the correct port.

2] Remove four screws securing fan to bottom of the equipment rack.

3] Remove fan.

b) Installation

1] Position fan to underside of equiment rack and secure with four screws.

2] Reconnect wire connector and air duct hoses (see “Figure 4” on page 212014).

NOTE: Cap any unused fan ports.

3] Reinstall MFD.

4) Remote Fan (Safe 328)

a) Removal

1] Disconnect air duct hoses and wire connector.

2] Remove three (3) nuts and washers securing fan to mounting bracket.

3] Remove fan.

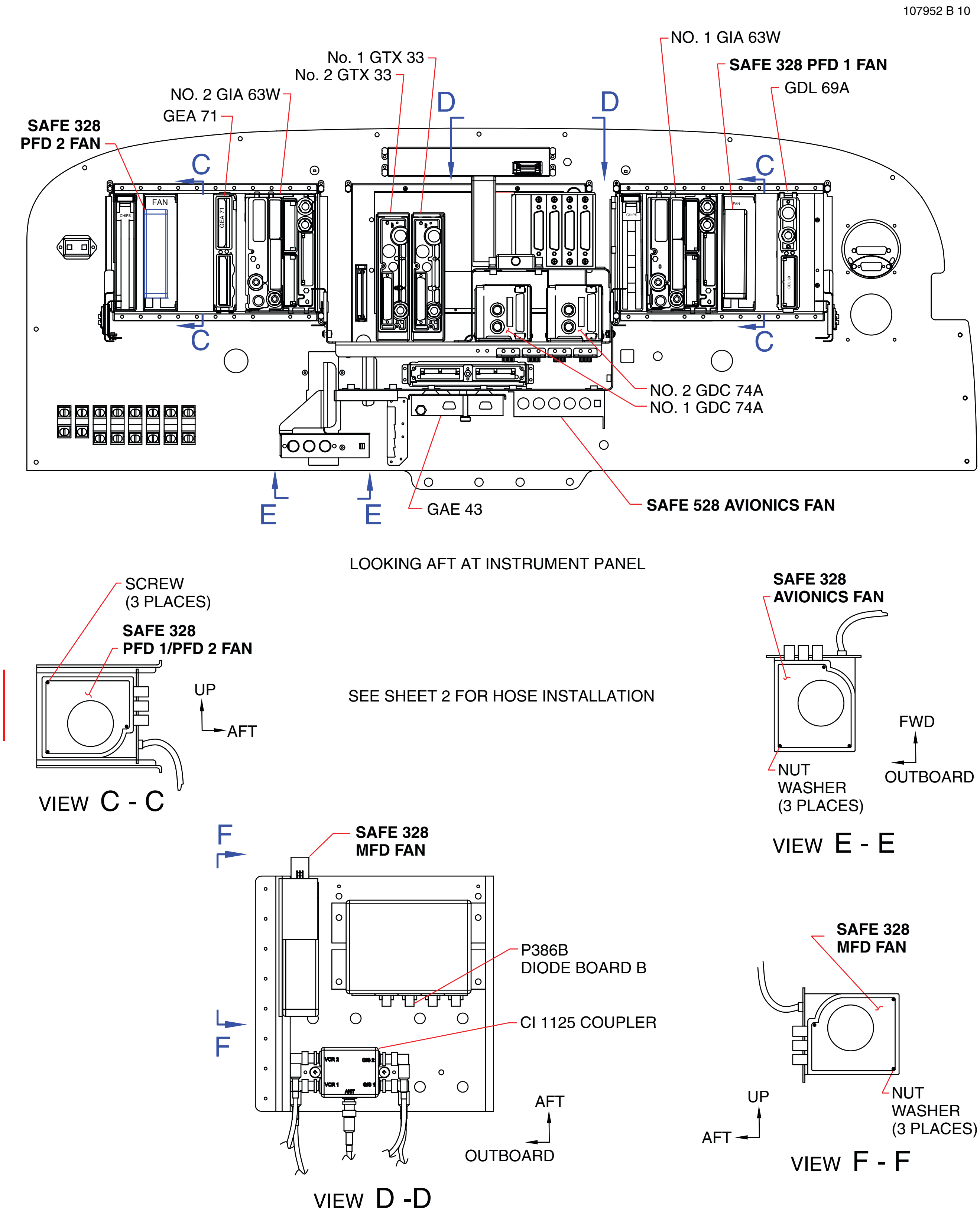
b) Installation

1] Position fan to underside of equiment rack and secure with four screws.

2] Reconnect wire connector and air duct hoses (see “Figure 4” on page 212014).

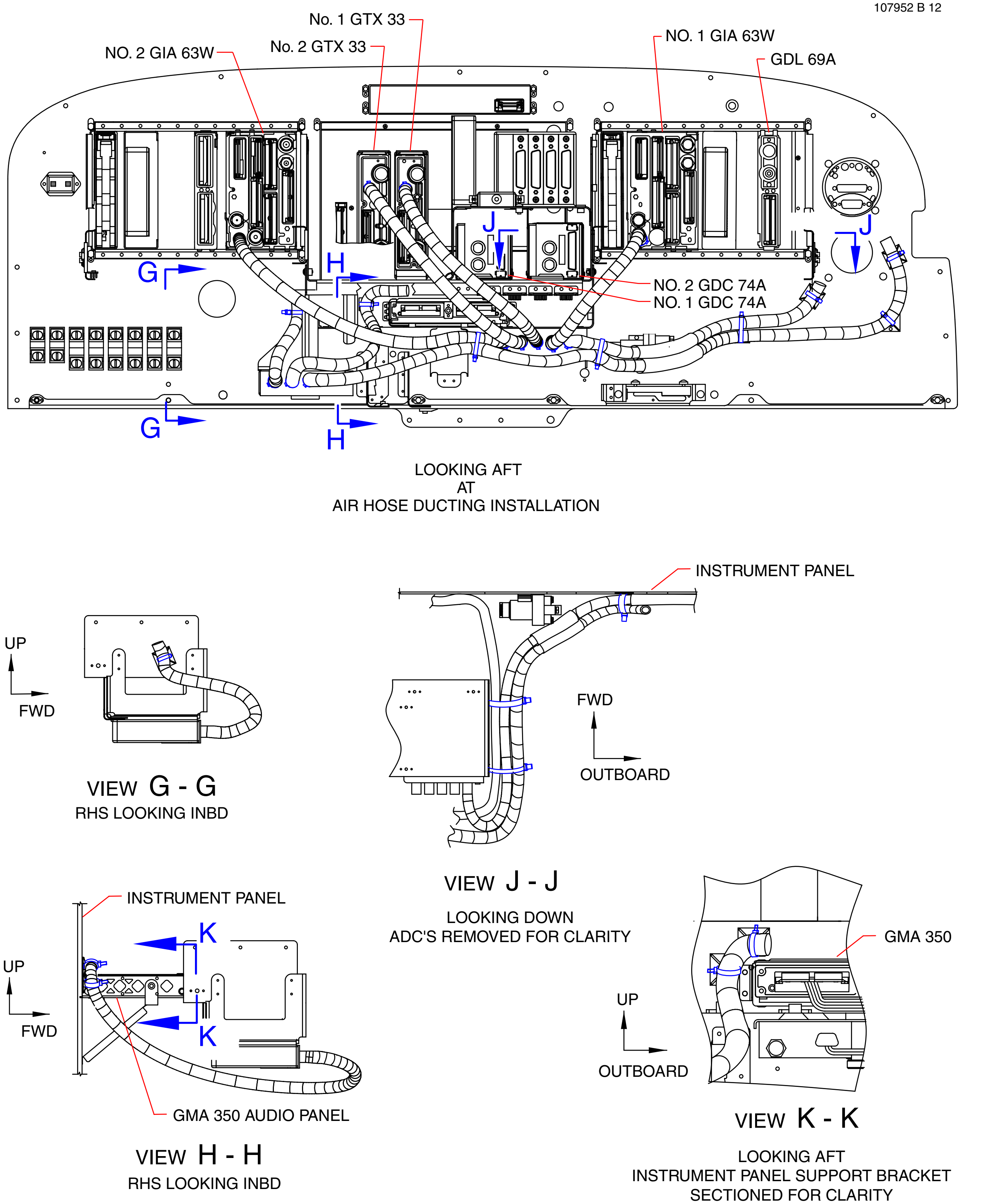
NOTE: Cap any unused fan ports.

3] Reinstall MFD.



[Effectivity](#)
4636633, 4636652 -
4636715, 4636717-4636719

Avionics Cooling (2014 - 2017)
Figure 4 (Sheet 1 of 3)



Avionics Cooling (2014 - 2017)
Figure 4 (Sheet 2 of 3)

Effectivity
4636633, 4636652 -
4636715, 4636717-4636719

(b) Aft Equipment Shelf Cooling Fan (Safe 328) (Optional)

The sixth (6th) fan (Safe 328) is mounted to a bracket on the right side of the lower aft equipment shelf and uses air duct hoses to provide cooling to the GTS 825 and other optional electronic equipment.

1) Removal

a) Remove cabin aft closeout panel.

b) Disconnect wire connector from fan.

c) Loosen clamp(s) and disconnect hose(s).

NOTE: Make note/tag each duct hose to enable reinstallation to the correct port.

d) Remove the three (3) screws securing fan to bracket.

e) Remove fan.

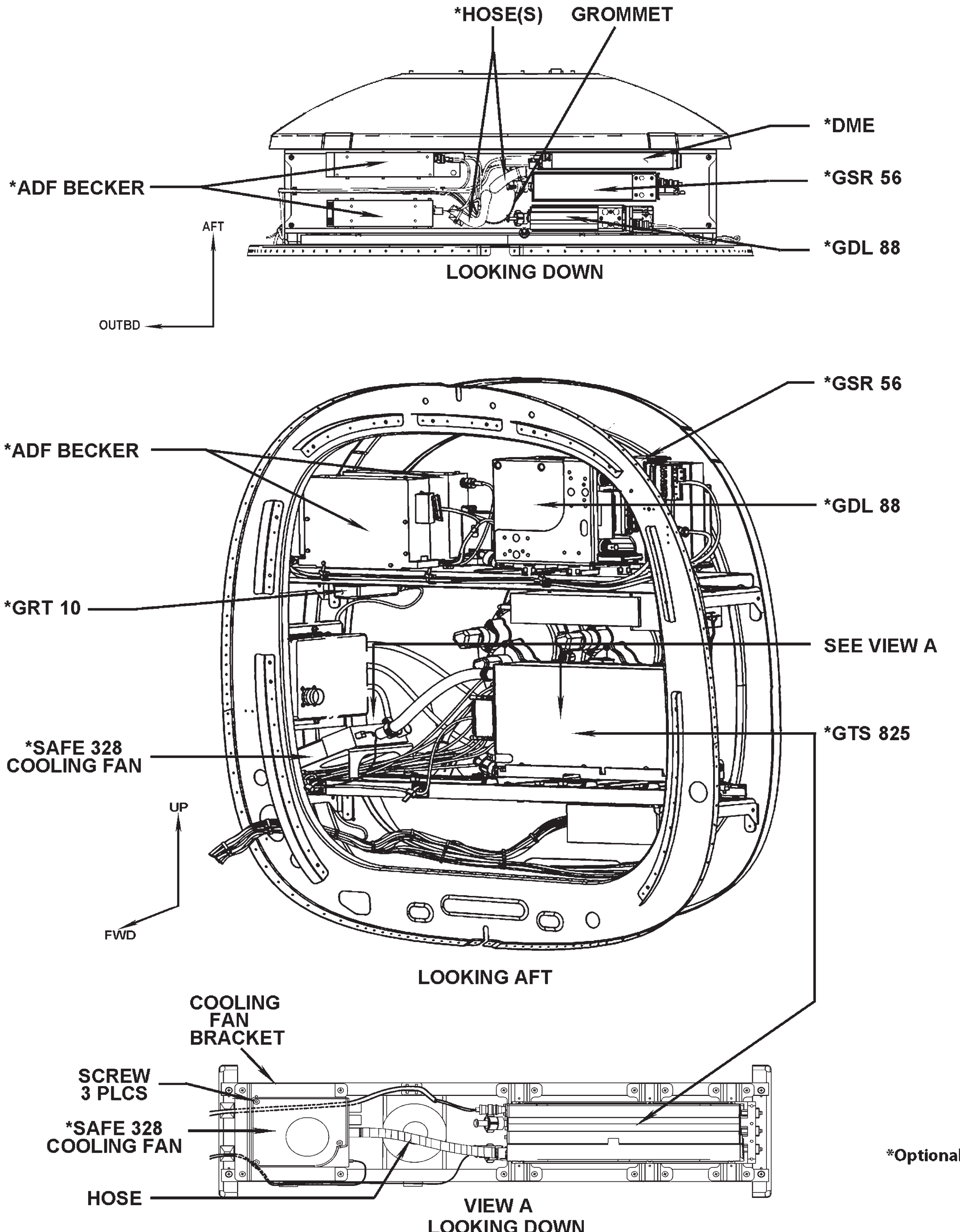
2) Installation

a) Align fan to mounting bracket, secure with three (3) screws.

b) Reconnect hose(s) and tighten clamp(s).

NOTE: Cap any unused fan ports.

c) Reconnect wire connector.



Avionics Cooling (2014 - 2017)
Figure 4 (Sheet 3 of 3)

Effectivity
4636633, 4636652 -
4636715, 4636717-4636719

(3) S/N's 4636716, 4636720 and up

See "Figure 5" on page 212019.

Six (6) cooling fans (blowers) are installed behind or under the instrument panel. Four (4) fans (blowers) are rack mounted one (1) fan (with base) behind each PFD and two (2) behind the MFD (mounted on shelf above GDC 72's), with one providing cooling using duct hoses for GIA #1, GTX and ADC. The remaining two (2) fans (blowers) are remote mounted under the equipment rack behind the MFD on each side of the GAE-43 and provide cooling using duct hoses for GIA #2, GTX, GMA 350C on one side and GIA #1 and the Aspen EBD on the other side.

(a) Rack Mounted Fans (4 ea.)

1) Removal

a) Remove the GDU. (See Components in 34-25-02).

b) Disconnect wire connector.

c) Remove the four (4) screws, nuts and washers.

d) Remove fan from rack and remove screws that attach fan to the mount (if applicable).

2) Installation

a) Reattach fan to rack mount and install mounting screws.

b) Place fan back in to the rack reinstall the screws securing the fan (with mount if applicable) to rack base.

c) Reconnect wire connector.

d) Reinstall GDU. (See Components in 34-25-02).

(b) Remote Fans (2 ea.)

1) Removal

a) Disconnect air duct hoses and wire connector (see "Figure 5").

NOTE: Make note/tag each duct hose to enable reinstallation to the correct port.

b) Remove four screws securing fan to bottom of the equipment rack.

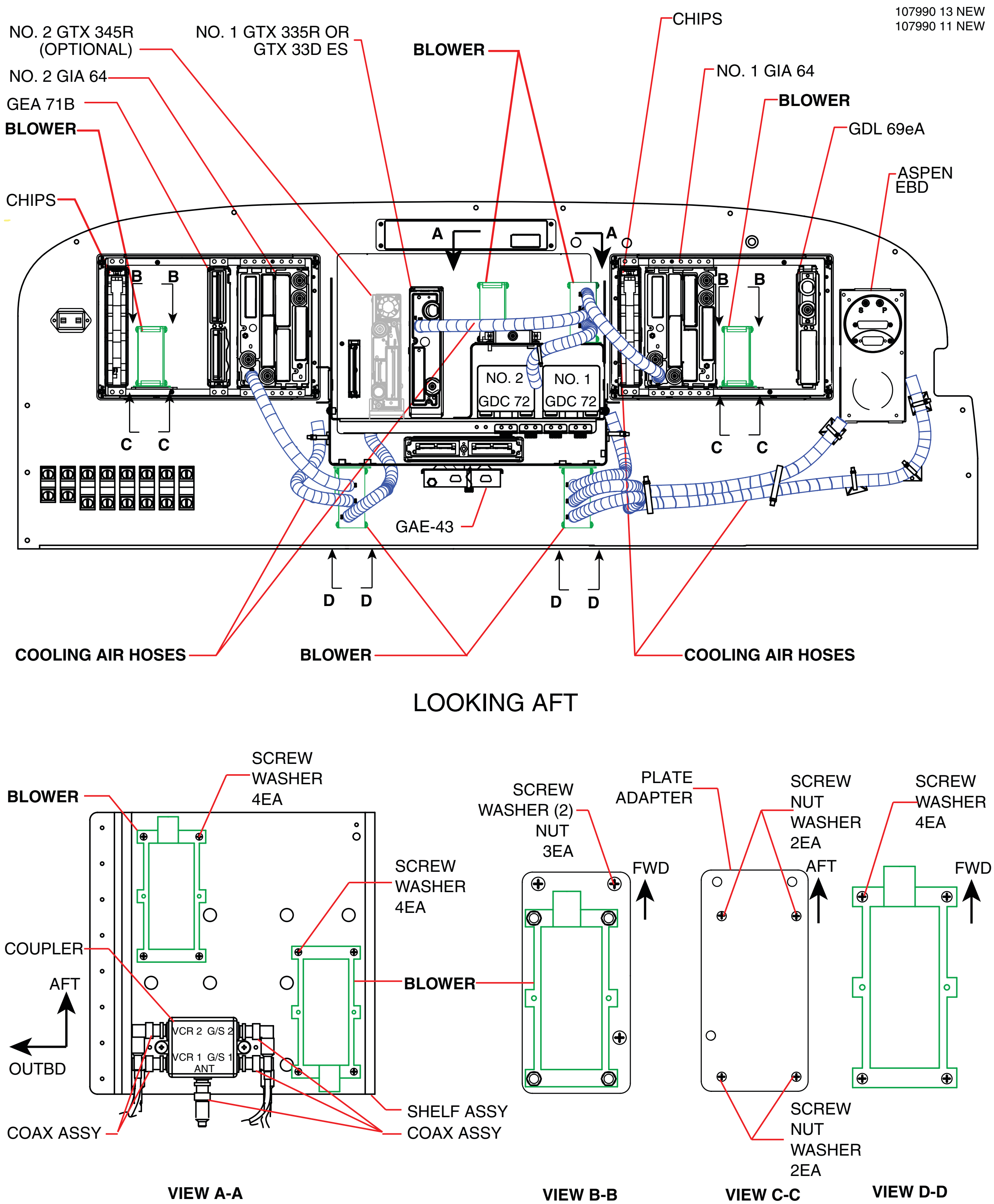
c) Remove fan.

2) Installation

a) Position fan to underside of equiment rack and secure with four screws.

b) Reconnect wire connector and air duct hoses (see "Figure 5").

NOTE: Cap any unused fan ports.



Avionics Cooling (2017 and up)
Figure 5

Effectivity
4636716, 4636720 and up

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PRESSURIZATION CONTROL (PA-46-350P ONLY)

These airplanes are equipped with two different pressurization systems. A conventional pneumatic system in early airplanes (1995–2014) and an electro-mechanical system that is digitally controlled in later airplanes (2015 and up).

1. Pneumatic System (S/N's 4636001– 4636651 less 4636633)

   A. Description

   The cabin pressurization and control system consists of an isobaric outflow valve, a safety outflow valve, cabin altitude and rate selector, electronically operated vacuum solenoid valve, surge tank, and associated interconnecting plumbing and wiring. Cabin altitude, differential pressure, and rate of change are displayed on a single three inch diameter indicator, or on the MFD in airplanes equipped with Garmin G1000 IAS. Should cabin pressure altitude exceed 10,000 feet:

   - the CABIN ALTITUDE annunciator will illuminate to warn the pilot;
   - except in Garmin G1000 equipped airplanes, where the red CABIN ALT 10000 CAS message will illuminate in the MFD to warn the pilot.

   Controls and switches needed to operate the cabin pressurization system are located on the lower section of the pilot's instrument panel and on the lower copilot's instrument panel. In addition to the CABIN PRESS and CABIN TEMP controls, are the cabin pressure and rate controller located:

   - in S/N's 4636001–4636020, just above the tie bus circuit breakers, and the CABIN PRESS DUMP/NORM switch located on the pilot's main switch panel.
   - in S/N's 4636021–4636374, just above the fuel selector control, and the CABIN PRESS DUMP/NORM switch located on the right overhead switch panel.
   - in airplanes equipped with Avidyne Entegra, to the left of the standby altimeter, and the CABIN PRESS DUMP/NORM switch located on the right overhead switch panel.
   - in airplanes equipped with Garmin G1000, below the standby airspeed indicator, and the CABIN PRESS DUMP/NORM switch located on the right overhead switch panel.

   For pressurized flight, set the cabin pressure controller at 500 feet above the airport pressure altitude, CABIN PRESS control knob full in and the CABIN PRESS DUMP/NORM switch to NORM. The rate of cabin ascent and descent change is controlled with the rate knob (left lower corner of the cabin pressure controller), and may be adjusted between approximately 200 and 2000 feet per minute, as desired. Setting the rate knob arrow to the 9 o'clock position provides a cabin altitude rate of change of approximately 500 feet per minute. This position gives a comfortable rate for normal operations.

   In S/N's 4636001–4636459, 4636461–4636462, and 4636481:

   Below/next to the cabin pressure controller (or pilot's PFD, if Avidyen Entegra equipped), a triple indicator simplifies monitoring the system's operation. The triple indicator displays the cabin altitude, cabin rate of change and the differential pressure between the cabin and the outside atmosphere. Maximum cabin differential pressure is 5.5 psi.

   A CABIN ALTITUDE warning light on the annunciator display warns the pilot when the cabin altitude is above 10,000 feet. Cabin pressure is automatically regulated to a maximum of 5.5 psi pressure differential. Should the cabin outflow valve malfunction, the cabin safety valve will maintain a maximum of 5.6 cabin differential pressure.

In airplanes equipped with Garmin G1000:

A CABIN ALT 10000 warning CAS message is illuminated when the cabin altitude is above 10,000 feet. Cabin pressure is automatically regulated to a maximum of 5.6 psi pressure differential. Should the cabin outflow valve malfunction, the cabin safety valve will maintain a maximum of 5.6 psi cabin differential pressure. Should the cabin differential pressure remain above 5.6 psid for greater than 30 seconds, or if it reaches 5.8 psid, a Master Warning will be triggered, with a red DIFF PSI indication (on the MFD) and a repeating aural chime.

The landing gear squat switch, on the left main landing gear, prevents the cabin from being pressurized while the airplane is on the ground.

The CABIN PRESS DUMP/NORM switch, when set to DUMP, electrically opens a solenoid valve allowing vacuum suction pressure to open the safety valve and rapidly dump cabin pressure to ambient pressure.

For unpressurized flight the CABIN PRESS control should be pulled fully out. Setting the CABIN PRESS/DUMP/NORM switch to DUMP will provide maximum airflow through the cabin. Cabin temperature will continue to be controlled by the CABIN TEMP control.

See Chapter 91 for electrical schematics.

B. Troubleshooting

See “Chart 1”.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING PRESSURIZATION SYSTEM (PNEUMATIC)

Trouble	Cause	Remedy
Cabin will not pressurize.	Blocked safety valve cabin air filter or orifice.	Replace filter and check orifice.
	Blocked controller cabin air filter with orifice.	Clean filter and check orifice on controller.
	Internal malfunction in the isobaric valve and defective absolute pressure regulator.	Remove and replace isobaric valve and absolute pressure regulator.
	Internal malfunction in the safety valve.	Remove and replace safety valve.
	Internal malfunction in the controller.	Remove and replace controller.
	Defective landing gear safety switch.	Remove and replace switch.
	Dump switch incorrectly positioned or defective.	Replace switch.
	Excessive leakage in pressurization ducts.	Locate leak and tighten connections.
	Excessive cabin leakage.	Locate and repair leak.
	Solenoid valve malfunctions in the open position.	Replace valve.
Cabin pressurizes to full positive differential pressure after takeoff.	Vacuum tube not connected to controller.	Connect vacuum tube.
	Malfunction in aircraft vacuum supply.	Check aircraft vacuum supply.
	Rupture in volume tank.	Replace tank.
	Internal malfunction in the outflow isobaric valve.	Remove and replace valve.
	Internal malfunction in the controller.	Remove and replace controller.
Cabin altitude decreases below selected altitude.	Low aircraft vacuum supply.	Check aircraft vacuum supply.
	Minor leak in tube between controller and volume tank or in volume tank.	Remove and replace tube or volume tank.
	Minor leak in tube between controller and isobaric valve or or in isobaric valve.	Remove and replace tube or isobaric valve.
	Minor leak in the controller.	Remove and replace controller.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING PRESSURIZATION SYSTEM (PNEUMATIC)

Trouble	Cause	Remedy
Minimum rates unbalanced. Down rate faster than up rate.	Minor leak in tube between controller and volume tank or in volume tank.	Remove or replace tube or volume tank.
	Minor leak in the controller.	Remove and replace controller.
Pressurized operation before takeoff and after landing.	Solenoid valve malfunction in the closed position.	Replace valve.
	Landing gear safety switch/ dump switch malfunction in the open position.	Remove and replace switch.
	Broken wiring to the vacuum solenoid valve.	Repair wiring.
	Loose or damaged pneumatic tube between vacuum solenoid valve and safety valve.	Check tube connections or replace damaged tube.
	Internal malfunction in the safety valve.	Remove and replace valve.
Cabin exceeds full positive differential calibrated setting.	True static atmosphere tube blocked or not connected on isobaric valve and safety valve.	Securely connect or remove blockage from the true static atmosphere tube on isobaric valve and safety valve.
	Loose or damaged pneumatic tubing from port “1” of safety and isobaric valve to atmosphere.	Repair or replace tubing.
Cabin climbs and descends at a fixed rate regardless of rate selection.	Internal malfunction in controller.	Remove and replace controller.
Cabin rate exceeds selected rate valve during aircraft climb to cruise altitude.	Malfunction in controller.	Remove and replace controller.
	Defective cabin rate of climb indicator.	Remove and replace indicator.
Cabin pressure rapidly increases or decreases with reselection of cabin altitude.	Malfunction in controller.	Remove and replace controller.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING PRESSURIZATION SYSTEM (PNEUMATIC)

Trouble	Cause	Remedy
Cabin altitude exceeds selected value.	Loss of airflow into cabin.	Repair bleed air supply.
	Internal malfunction in isobaric valve.	Remove and replace valve.
	Internal malfunction in safety valve.	Remove and replace valve.
	Internal malfunction in controller.	Remove and replace controller.
	Excessive cabin leak rate.	Seal leaks.
Cabin pressure will not maintain control setting.	Defective outflow valve.	Remove and replace valve.
	Defective safety valve.	Remove and replace valve.
	Defective cabin altitude controller.	Remove and replace cabin altitude controller.
	Pressurization duct leakage.	Locate leak and tighten connection.
	Cabin leakage.	Locate and repair leak.
	Defective cabin altitude pressure transducer.	Remove and replace transducer.
	Foreign matter on isobaric outflow control valve or safety valve seats.	Clean valve seats.
	Leak in outflow isobaric valve control line.	Locate leak and tighten connection.
Cabin pressure excessively high.	Isobaric outflow valve static line clogged.	Clean static line.
	Defective cabin over pressure switch.	Remove and replace switch.
	Safety valve static vent clogged.	Clean vent line.
	Defective safety valve.	Remove and replace valve.

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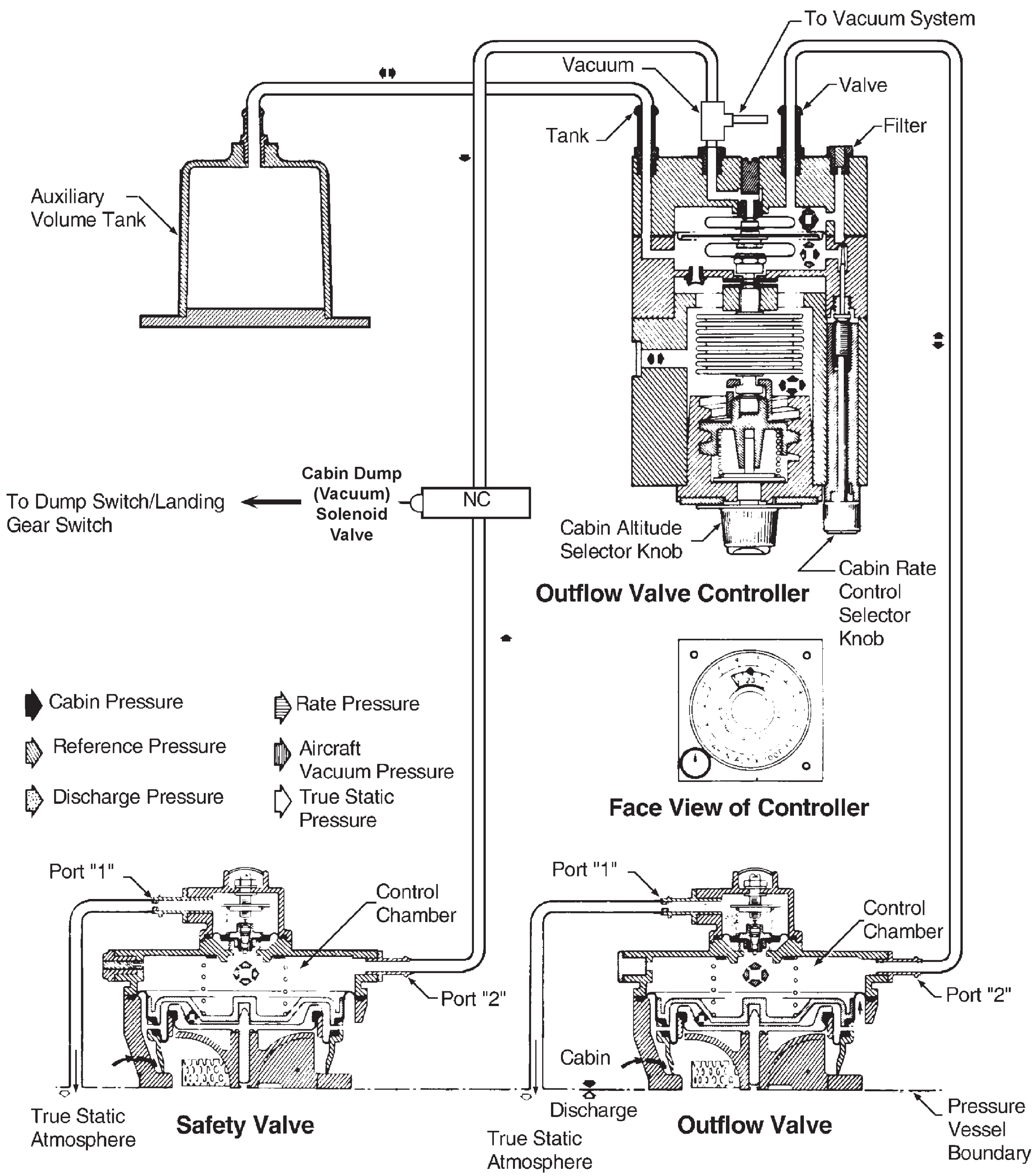


Cabin Pressure Control System
Figure 1

C. Components

(1) Cabin Outflow and Safety Valves

See “Figure 2”.

(a) Description

1) Safety Valve

The safety valve consists of two sections, a head section and a base section. The head section contains port “2” depressurization connection and a cabin air filter with orifice. Mounted on the head is a differential control which contains port “1” (true static atmosphere connection with orifice). The base assembly encloses a poppet outflow valve which seats on a surface of the base assembly when in the closed position.

2) Cabin Outflow Valve.

The outflow valve consists of two sections, a head section and a base section. The head section contains port “2” (controller connection) and a plugged cabin air port. Mounted on the head is the differential control, which contains port “1” (true static atmosphere connection with orifice). The base assembly encloses a poppet outflow valve which seats on a surface of the base assembly when in the closed position.

(b) Removal

1) Disconnect the two lines from the valve being removed.

2) Remove the three nuts which secure the valve to the aft pressure bulkhead.

3) Remove the valve from the pressure bulkhead.

(c) Installation

1) When reinstalling outflow and/or safety valve, the gasket dimension shown in Detail A of “Figure 2” must be met to ensure proper sealing. To provide proper clearance between nuts and gasket face, adjust for clearance as follows:

a) Remove nuts using a deep socket and remove enough shims to meet dimensional requirements. A minimum of one shim is required each place for installation of nut.

b) Install nuts and tighten.

c) Apply a drop of locking compound, ASTM-D-22473, to the top of nuts between nuts and studs.

2) Position the valve on the rear pressure bulkhead.

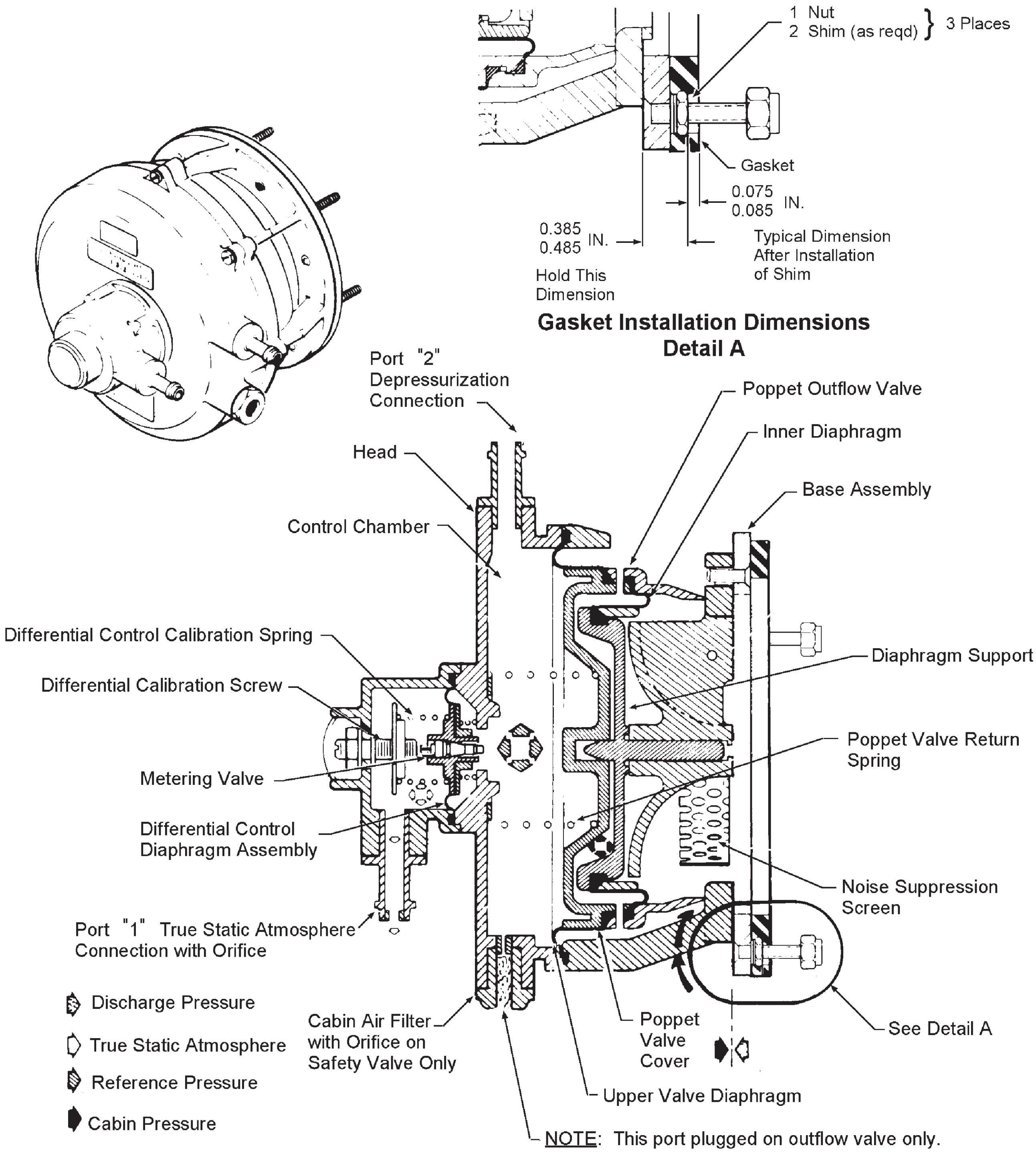
3) Install the three nuts which hold the valve to the aft pressure bulkhead. Torque nuts to 20 in.-lbs.

4) Reconnect the two lines to the valve.

(d) Servicing

Routine maintenance of the valves is limited to the replacement of the filter in the safety valve and cleaning of the seats in both valves.

Using Joy detergent or isopropyl alcohol, clean the seats of either valve thoroughly.



Safety and Outflow Valves
Figure 2

(2) Cabin Pressure Controller

See "Figure 3".

(a) Description

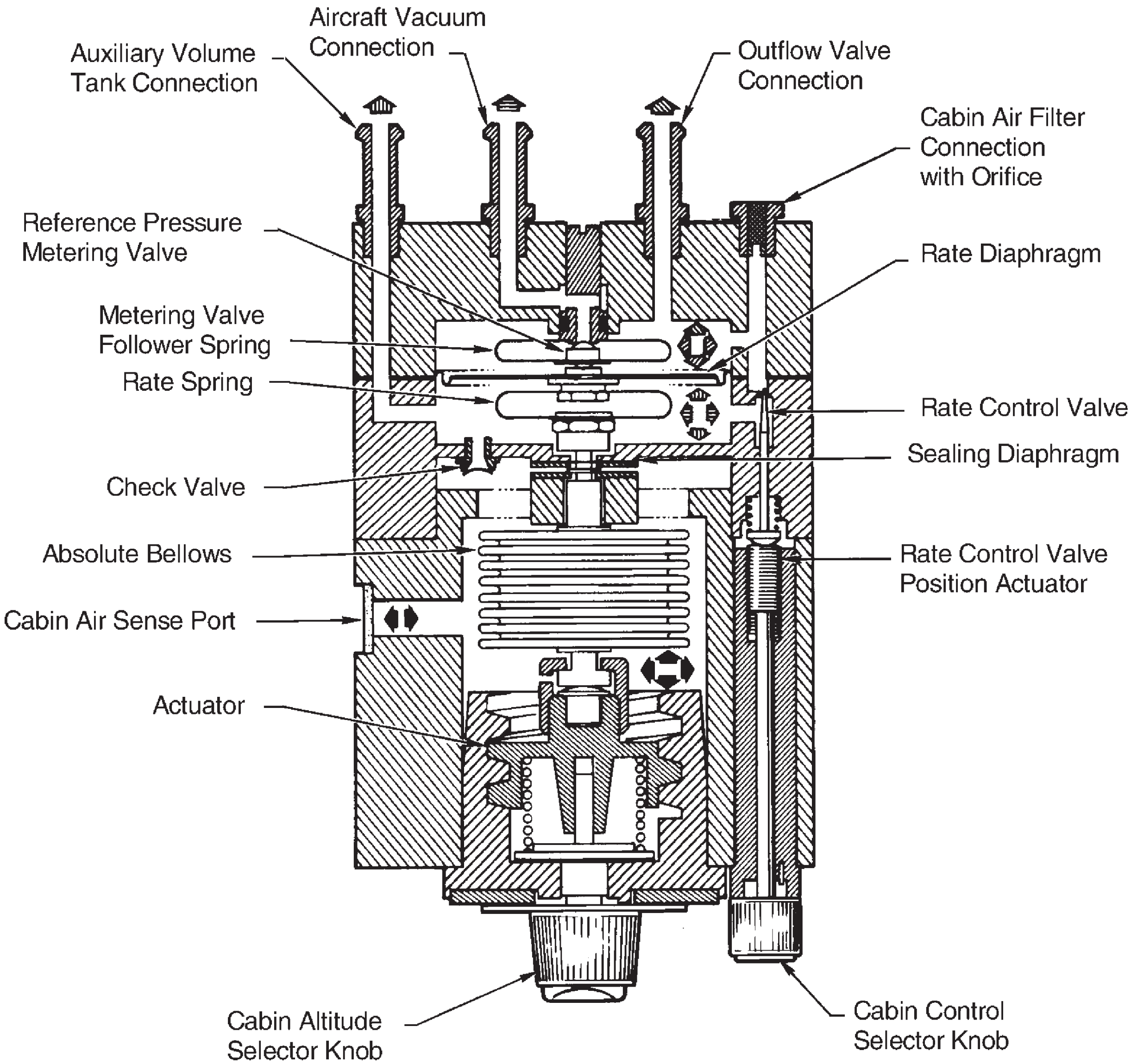
The cabin pressure controller is located on the lower left of the instrument panel. The body of the controller is composed of three elements. The lower element forms a chamber which is open to cabin pressure and houses the absolute bellows. It also is the face of the controller, and has the cabin altitude selector knob and the cabin rate control knob. The middle element, when assembled to the lower element, completes the cabin pressure chamber and seals the cabin pressure chamber from the rest of the controller with a sealing diaphragm. The opposite side of the middle element forms a rate pressure chamber which houses the rate spring. The upper element, when assembled to the rate diaphragm and other two elements, forms a reference pressure chamber which houses the reference pressure metering valve and metering valve follower spring. This element also contains the various air connections for installation of the controller into the cabin pressure control system.

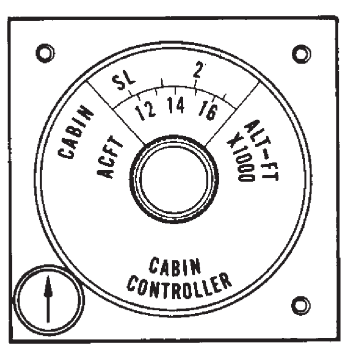
(b) Operational Check

(PIR-Garrett Report No. 4-360.)

This check verifies minimum ▲P operation, cabin altitude control, and cabin pressure rate-of-change control are operational.

1) Close and secure cabin door.
2) Rotate cabin rate control selector knob to the 9 o'clock position.
3) Rotate cabin altitude selector knob until CABIN ALT indicates approximately 500 feet above field altitude.
4) Set brakes, start engine, and establish a steady cabin air inflow.
5) Close the squat switch by placing a wedge or block of wood between the striker plate and switch (see 32-60-00). Note that safety valve closes and outflow valve remains open.
6) Rotate cabin altitude selector knob counterclockwise until CABIN ALT indicates approximately 1500 feet below field altitude or until it reaches its stop. Cabin pressure will increase on rate-of-change operation.
7) Rotate cabin rate control selector knob counterclockwise and note a reduction in cabin pressurization rate-of-change.
8) Rotate cabin rate control selector knob clockwise and note an increase in cabin pressurization rate-of-change. Return selector knob to the 9 o'clock position.
9) Allow cabin pressure to stabilize at the selected value.
10) Rotate cabin altitude selector knob until CABIN ALT indicates approximately 500 feet above field altitude.
11) Rotate cabin rate control selector knob counterclockwise and note a reduction in cabin pressurization rate-of-change.
12) Rotate cabin rate control selector knob clockwise and note an increase in cabin pressurization rate-of-change; then, allow cabin pressure to return to field altitude and outflow valve to move to the full-open position.
13) Open the squat switch by removing the wedge or block of wood holding the switch closed. Note that safety valve also fully opens.
14) Shutdown engine.





Face View of Controller

Cabin Pressure Controller
Figure 3

(3) Auxiliary Volume (Surge) Tank

See “Figure 1” on page 21307.

The tank is mounted to a longeron in the left side of the fuselage between F.S. 100.00–108.00 at W.L. 76.15. This small sealed chamber is located close to the controller; and when connected to the controller, provides additional volume to the rate pressure chamber in the controller, thus providing greater accuracy of the cabin rate-of-control.

(4) Vacuum Relief (Cabin Dump) Solenoid Valve.

(a) Description

The solenoid valve is mounted on the left-hand side of the F.S. 108.00 bulkhead at W.L. 57.00. It is located in the control line between the cabin pressure controller vacuum port and the safety valve. This valve is normally closed. When energized, vacuum is applied to the diaphragm of the safety valve, holding it open. The vacuum relief solenoid valve is open under either of the following conditions:

1) On the ground with the landing gear squat switch energized.

2) In flight whenever the cabin dump switch is in the DUMP position.

(b) Removal

1) Gain access to the solenoid valve.

2) Disconnect the lines from the valve.

3) Remove the screws and washers that attach the valve to the bulkhead. Remove the valve.

(c) Installation

1) Secure the valve to the bulkhead with the screws and washers.

2) Reconnect the lines to the valve.

3) Tighten the nuts finger tight plus one and one-half turns.

(5) Vacuum Regulator

Vacuum to operate the safety valve and controller is obtained from the vacuum system which utilizes its own regulator. The regulator, set to provide a nominal suction of 5.0 inches of mercury (relative to cabin pressure), is incorporated in the system between the cabin and the ejector. See 37-10-00 for additional information.

(6) Cabin Pressure Transducer (with Garmin G1000 only)

In airplanes equipped with Garmin G1000 only, a Garmin GAE 43 (Sandia SAE 5-35) altitude encoder is used as the cabin pressure transducer.

The cabin pressure transducer is a small independent sensor module that provides cabin pressure altitude information for aircraft cockpit display. It’s mounted to the underside of the equipment shelf located forward of the MFD. It feeds cabin pressure altitude data to the GIA 63W. See Components in 34-25-01 for removal, installation, and calibration.

(7) Bleed Air Dump Valve

The bleed air dump valve is just forward of the firewall (F.S. 79.00) and above the bleed air crossover duct assembly. It is actuated by the cabin pressure control knob on the instrument panel.

(a) Removal

1) Disconnect control cable from top of bleed dump valve.

2) Remove bolts, nuts and washers that secure the bleed dump valve to the duct and exhaust duct/cable support.

3) Remove bleed dump valve.

(b) Installation

1) Position bleed dump valve in place.

2) Ensure that gaskets are installed between bleed dump valve and ducts.

3) Install bolts (heads pointing inboard) through duct flange, bleed dump valve, control cable support and exhaust duct flange. Secure bolts with nuts and washers. Torque the nuts from 10–12 in. lbs.

4) Attach control cable rod end to dump valve arm.

(8) Rigging Bleed Air and Bleed Air Condition System

(a) Rig cabin pressure system as follows:

1) Push cabin pressure control knob into the instrument panel.

a) Verify that bleed dump valve is in closed position.

b) Verify that air selector valve and bleed air shutoff valve are in open position.

2) Pull cabin pressure control knob out from instrument panel.

a) Verify that bleed dump valve is in open position.

b) Verify that air selector valve and bleed air shutoff valve are in closed position.

(b) In S/Ns 4636001 thru 4636447, rig cabin air temperature system as follows:

1) Push control cable knob into the instrument panel.

2) Verify that the cabin temperature control valve is open for ram air flow and closed to the heat muff hot air flow.

(c) In S/Ns 4636448 and up, rig cabin air temperature system as follows:

1) Push control cable knob into the instrument panel.

2) Adjust control cable assembly rod end fitting in or out to ensure that the air modulation valve is fully open for ram air flow and closed to the heat muff hot air flow.

3) Ensure sufficient thread engagement into the rod end fitting by inspection of witness hole in rod end. Add or remove one spacer, if required, to achieve adjustment.

2. Electro-Mechanical System (S/N's 4636633, 4636652 and up)

A. Description

See "Figure 4".

The cabin pressurization and control system is comprised of two identical outflow valves, a controller, a cabin pressure transducer, and associated interconnecting plumbing and wiring. The controller is mounted on a shelf in the rear of the airplane near the aft pressure bulkhead. Both outflow valves are mounted directly on the aft pressure bulkhead facing into the cabin. The cabin pressure transducer is mounted in the cockpit near the instrument panel.

Cabin pressure is controlled by two electromechanical outflow valves located on the aft pressure bulkhead. These valves provide identical functions of maintaining a desired pressurization schedule during all phases of flight without exceeding the maximum differential pressure of 5.5 +/-0.1 psi. If the cabin pressure control system develops a communications fault during flight:

- A CPCS FAULT caution CAS message is posted.
- The landing field elevation and weight on wheels sensors are lost, causing the cabin to remain pressurized after landing. Accordingly, the pilot must verify that the cabin is fully depressurized prior to landing.

If the cabin pressure control system develops a communications failure on the ground, a CPCS FAIL caution CAS message is posted. The outflow valves will remain open and the cabin will not pressurize. If a CPCS FAIL caution CAS message is experienced during flight, the outflow valves will close, allowing the system to continue to operate in a degraded mode.

The cabin pressurization system controls, switches, and displays are as follows:

- CABIN PRESS push-pull knob;
- CAB PRES DUMP / NORM switch ;
- Cabin Altitude, Differential Pressure, and Rate of Climb EIS indications on MFD; and
- Destination Field Elevation (DEST ELV) in TMR/REF window on PFD.

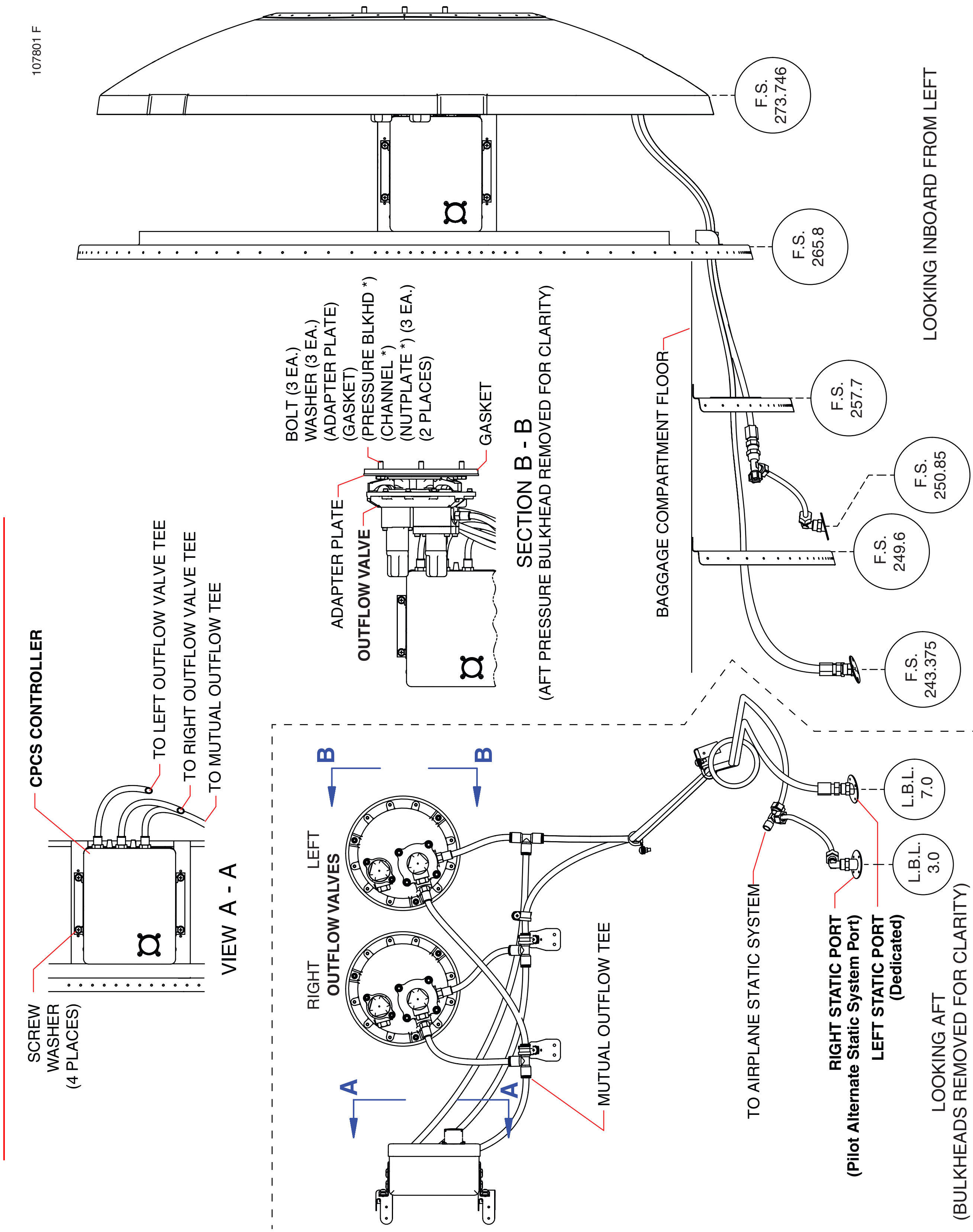
The only action required by the pilot during normal operation is to input the destination airport elevation by selecting the TMR/REF softkey on either PFD and entering the destination airport elevation in the DEST ELV field. If a new destination elevation is not entered, the last value entered will be used, which could result in the airplane landing while still pressurized.

The controller automatically schedules cabin pressure from takeoff to landing. If the pilot changes his destination in-flight, the landing field elevation information must be manually updated in the PFD. If communication fails in flight, the controller will use the last field elevation that was output. In flight, the cabin altitude and rate are automatically scheduled to minimize the pressure changes experienced by the passengers and crew. During descent, the controller automatically schedules a comfortable cabin altitude rate to allow the cabin to reach landing field pressure elevation upon landing. After landing, the controller ensures that the cabin remains safely depressurized.

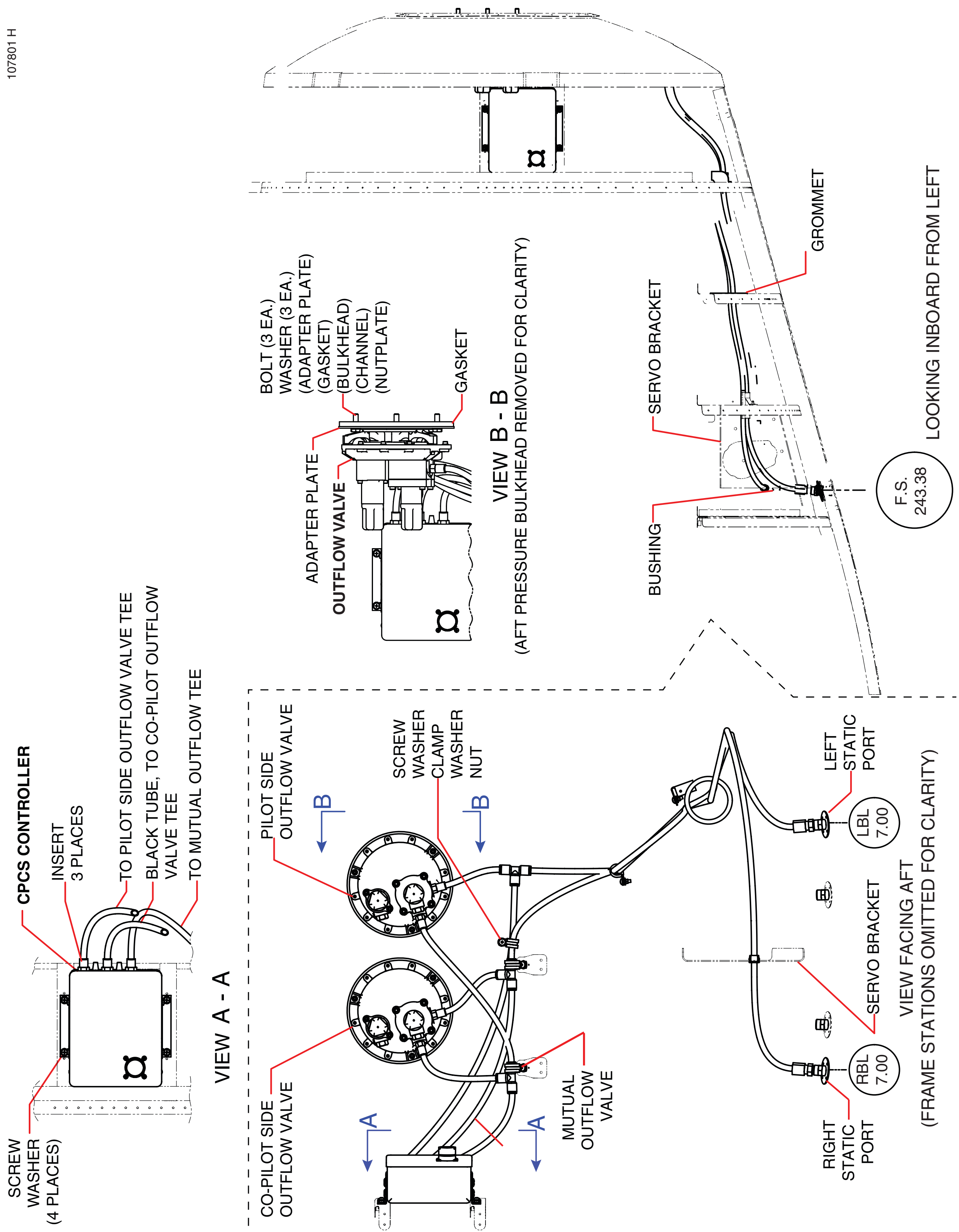
Cabin altitude, rate, and differential information are sensed by the cabin pressure transducer and are presented to the pilot on the avionics displays.

A CABIN ALT 10K warning CAS message is illuminated when the cabin altitude is above 10,000 feet. Cabin pressure is automatically regulated to a maximum of 5.6 psi pressure differential. Should the cabin differential pressure remain above 5.6 psid for greater than 30 seconds, or if it reaches 5.8 psid, a Master Warning will be triggered, with a red DIFF PSI indication (on the MFD) and a repeating aural chime.

When the CABIN PRESSURE DUMP switch is set to DUMP, the controller sets both outflow valves to full open and the valves remain open until the switch is disengaged or the maximum cabin altitude is reached.



Electro-Mechanical Cabin Pressurization System
Figure 4 (Sheet 1 of 2)



Electro-Mechanical Cabin Pressurization System
Figure 4 (Sheet 2 of 2)

For unpressurized flight the CABIN PRESS control should be pulled fully out. Setting the CABIN PRESS/DUMP/NORM switch to DUMP will provide maximum airflow through the cabin. Cabin temperature will continue to be controlled by the CABIN TEMP control.

Refer to Chapter 91 of this manual for electrical schematics of the Cabin Pressure Controller System.

B. Troubleshooting

(1) See “Chart 2” on page 213018.

(2) CPCS Lines Leak Test

See “Figure 4” on page 213015.

Troubles in the CPCS can frequently be traced to leaks in the control and/or static lines. Before replacing the cabin pressure controller or an outflow valve, conduct the following checks:

(a) Attach a static test device to one of the CPCS dedicated static ports.

(b) Increase altitude to 1,000 feet. Leak rate shall be less than 100 ft/min.

NOTE: If the leak rate exceeds 100 ft/min, check for leaks in the static lines between the static port, outflow valve and cabin pressure controller. Repair any leaks found and begin the test again.

(c) Increase altitude to 15,000 feet at a minimum climb rate of 3,000ft/min. This will allow the max differential pressure valve inside the outflow valve to actuate, allowing the pressure in the lines between the controller, outflow valves, and the static port to equalize. Decrease the altitude to 12,700 feet. Leak rate shall be less than 254 ft/min.

NOTE: It may appear as if there is a leak at approximately 5.5 psid. This is normal, continue to bring the altitude up to at least 1,000 feet higher and the apparent leak should disappear. Continue to 15,000 feet.

(d) If a leak exceeds the above tolerance, check fixture installation, plumbing and fittings. Repair any leaks found and repeat system checks above.

NOTE: If a leak exceeds the tolerance at 12,700 feet but passes at 1,000 feet, check for leaks in control ines (Pm line, see “Figure 6” on page 213023) between the cabin pressure controller and the outflow valves. It is also possible that the leak is in the outflow valve diaphragm or internal to the cabin pressure controller.

(e) Repeat steps (a) through (d) for the other dedicated CPCS static port, if installed.

CHART 2
TROUBLESHOOTING - PRESSURIZATION SYSTEM (ELECTRO-MECHANICAL)

Trouble	Cause	Remedy
Cabin will not pressurize.	Leak in CPCS static or control lines.	Locate and repair leak.
	Internal malfunction in an outflow valve.	Remove and replace defective outflow valve.
	Internal malfunction in the controller.	Remove and replace controller.
	Defective landing gear safety switch.	Remove and replace switch.
	Dump switch incorrectly positioned or defective.	Replace switch.
	Excessive leakage in pressurization ducts.	Locate leak and tighten connections.
	Excessive cabin leakage.	Locate and repair leak.
Cabin pressurizes to full positive differential pressure after takeoff.	Leak in CPCS static or control lines.	Locate and repair leak.
	Internal malfunction in the controller.	Remove and replace controller.
Pressurized operation before takeoff and after landing.	Leak in CPCS static or control lines.	Locate and repair leak.
	Defective landing gear	Remove and replace switch. safety switch.
Cabin pressure will not maintain control setting.	Leak in CPCS static or control lines.	Locate and repair leak.
	Defective outflow valve. Defective controller.	Remove and replace valve. Remove and replace controller.
	Pressurization duct leakage.	Locate leak and tighten connection.
	Excessive cabin leakage.	Locate and repair leak.
	Foreign matter on outflow valve seats.	Clean valve seats.

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C. Components

(1) Cabin Outflow Valve

See “Figure 4” on page 213015 and “Figure 5”.

(a) Description

The pressurization system includes two identical outflow valves. The valves pneumatically regulate the flow of exhaust air from the cabin, and require no electrical power to operate.

Each outflow valve features an independent maximum differential pressure safety valve, and a maximum altitude safety valve. Isolation is provided between outflow valves to prevent a single fault from disabling both maximum differential pressure safety valves. This is implemented via a 0.033 diameter restrictor orifice at each of the outflow valve common ports. Together, these outflow valves meet all applicable regulations regarding maximum and negative differential pressure; no additional safety valves are required. The design of the outflow valve inherently prevents negative differential pressure; when the outside pressure is greater than the cabin pressure, the diaphragm lifts off the grill without friction or spring bias, equalizing cabin and outside pressure.

(b) Removal

1) Disconnect the two pneumatic lines from the valve being removed.

2) Remove the three bolts and washers which secure the valve to the aft pressure bulkhead.

3) Remove the valve from the pressure bulkhead.

4) If replacing valve, remove six screws and remove adapter plate. Retain screws and adapter plate for new valve.

NOTE: Gasket is attached to pressure bulkhead with low adhesion sealant. Gaskets in good condition may be reused.

(c) Installation

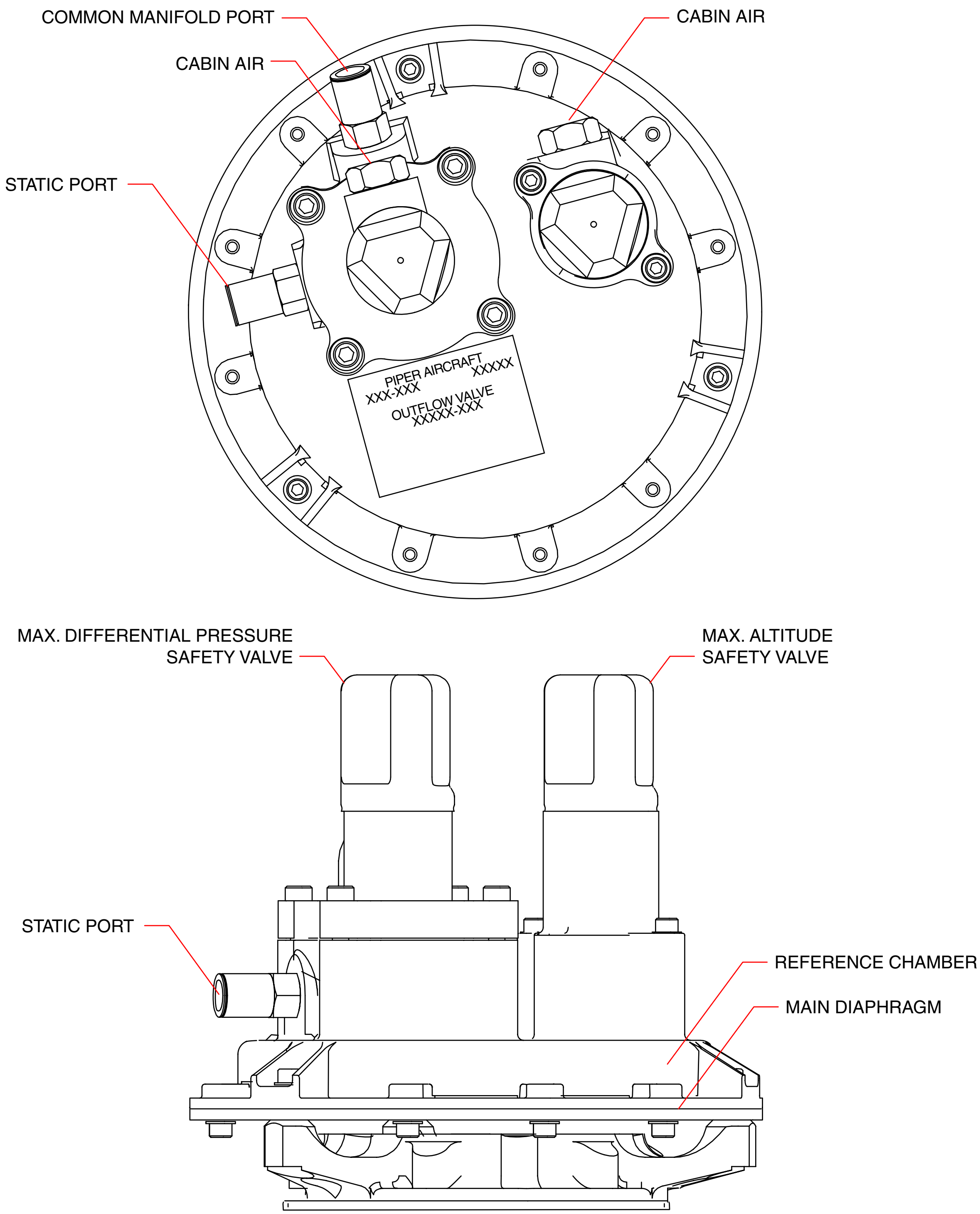
NOTE: If gasket requires replacement, remove old gasket and sealant. Clean aft pressure bulkhead surface with acetone. Attach new gasket to aft pressure bulkhead with AMS-3267/4 Class B-2 or AMS-3284 Type 2 Class B sealant.

1) If installing new outflow valve, attach adapter plate to valve with six screws.

2) Position the valve on the rear pressure bulkhead, aligning bolt holes in valve with bolt holes in bulkhead.

3) Install the three bolts and washers which hold the valve to the aft pressure bulkhead and tighten.

4) Reconnect the two pneumatic lines to the valve.



Outflow Valve
Figure 5

(2) Cabin Pressure Controller

See “Figure 4” on page 213015 and “Figure 6”.

(a) Description

The controller is mounted to brackets on the right side of the airplane between the F.S 265.8–273.746 bulkheads. The controller is designed to automatically regulate cabin pressure altitude and cabin pressure altitude rate of change for all phases of flight. All the electronics to operate the CPCS are packaged in the controller. The controller also contains integral vacuum pumps, solenoids, and integral pressure transducers to measure cabin pressure and cabin-to-static differential pressure. The measured cabin pressure altitude, cabin altitude rate, and cabin differential pressure allows for a backup mode to be used in the event of a communication failure between the aircraft and the controller.

The cabin can be manually dumped in the event of an emergency by use of a guarded switch on the instrument panel. Once the switch is engaged the controller sets both valves to full open and the valves remain open until the switch is disengaged or the max cabin altitude is reached. The controller also includes features like Power on Built-In-Test (BIT) and Continuous BIT.

The controller receives two discrete inputs: Squat and Take-off Power. The input signals may be hardwired or sent over the ARINC interface. The function of each discrete is described below:

1) Squat (WOW): This discrete lets the controller know if the aircraft is on the ground or in the air. It is used for pre-pressurization and for cabin dump upon landing.

2) Take-off Power (Throttle): When the throttle is set at approximately 80% or more power on the ground, the controller will pressurize the cabin up to 200 feet below takeoff altitude (i.e., field elevation) to prevent any pressure spikes during takeoff. This discrete is from the avionics suite.

To save weight and simplify the design, two internal vacuum pumps are integrated into the controller. These pumps are needed during ground testing of the system, during take-off, landing and decompression when the differential pressure between static and cabin air is below 0.2 in Hg. When the differential pressure between static and cabin air is greater than 0.2 in Hg, the vacuum pumps are shutoff, which increases their service life. These pumps also eliminate the need to interface with the engine bleed air system to create a vacuum, which eliminates water and other form of contamination from the engine.

There are also two solenoids located inside the controller. One controls the dive function of the system while the second one controls the climb function. The solenoids are used to control the pressure in the reference chamber of the outflow valves, which sets their operating points. The modulation of the solenoids as commanded by the microprocessor, assures a smooth controlled change to the operating point of the outflow valves.

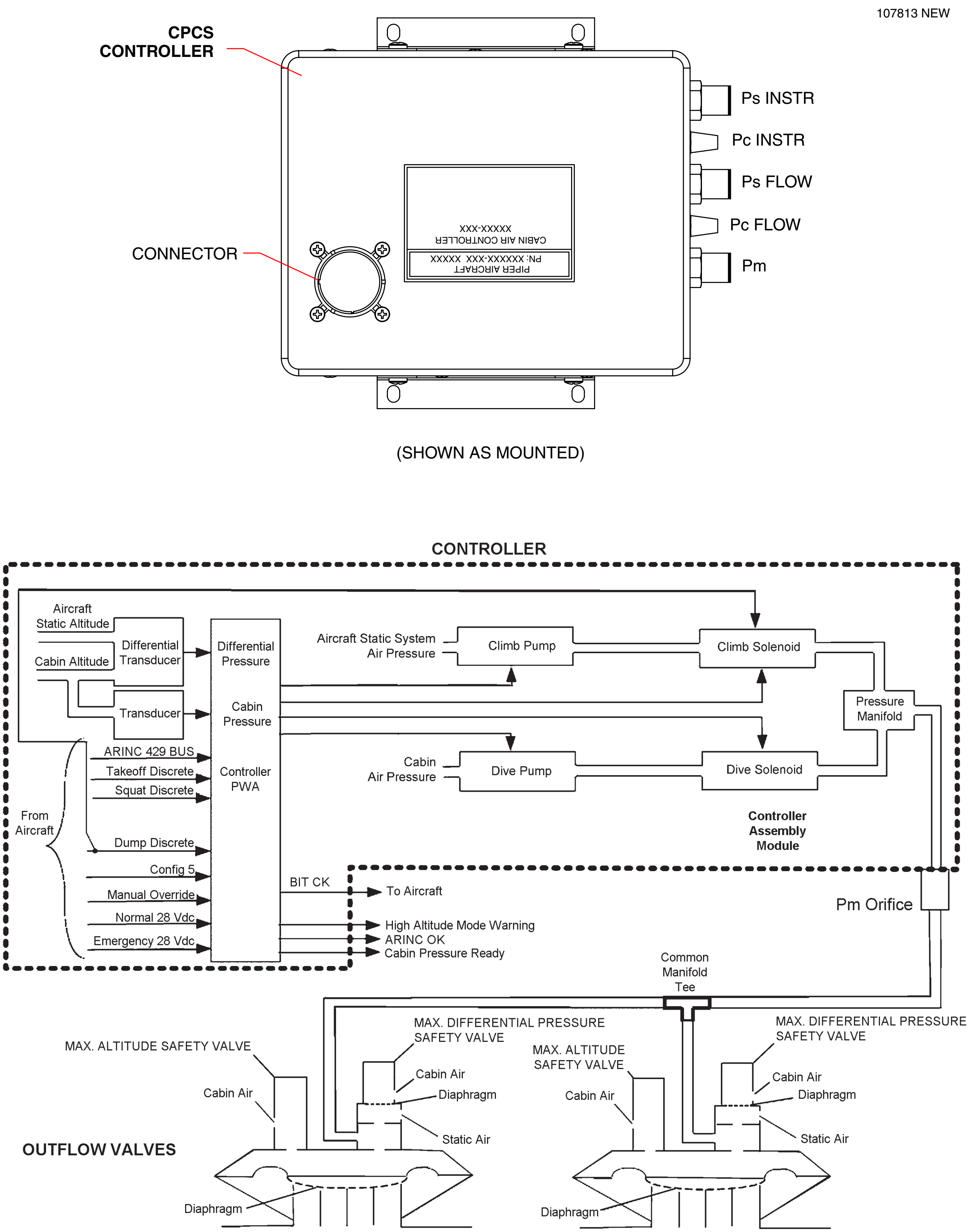
(b) Removal

1) Turn off master switch. Pull CABIN PRESS CONTROL and CABIN PRESS DUMP circuit breakers.

2) Disconnect connector from the controller.

3) Disconnect pneumatic lines from controller pneumatic ports.

4) Remove four (4) screws and washers securing controller to mounting brackets and remove controller.



Electro-Mechanical Cabin Pressurization Controller
Figure 6

(c) Installation

1) Position controller to mounting brackets with pneumatic ports aft.

2) Secure controller to mounting brackets with screws and washers (4 ea.).

3) Connect pneumatic lines to controller as shown in “Figure 4” on page 213015.

4) Connect connector.

5) Push in CABIN PRESS CONTROL and CABIN PRESS DUMP circuit breakers.

(3) Cabin Pressure Transducer

The cabin pressure transducer is a small independent sensor module that provides cabin pressure altitude information for aircraft cockpit display. A Garmin GAE 43 (Sandia SAE 5-35) altitude encoder is used as the cabin pressure transducer. It’s mounted behind the instrument panel lower center section underneath the GMA 350/GMA 350c audio panel. It feeds cabin pressure altitude data to the GIA 63W/GIA 64. See 34-25-01 (G1000) or 34-25-02 (G1000 NXi), Components, GAE 43 Cabin Altitude Encoder for removal, installation, and calibration.

3. Pressurization System Test

A. Pneumatic Pressurization System (S/N's 4636001–4636651 less 4636633)

(PIR-PPS60144-2, Rev. I.)

WARNING: DO NOT APPLY ELECTRICAL POWER ON BUS UNLESS AIRCRAFT IS ON JACKS AND STRUTS ARE FULLY EXTENDED.

(1) Equipment Required

Pressurization Test Cart (commercially available)

(2) Procedure

NOTE: Remove cowling before beginning.

(a) Plug cabin altimeter indicator (3-in-1 indicator) sensing port (marked "C.P.").

(b) Disconnect vacuum line at the tee on the rear of the cabin pressurization controller. Cap tee and plug vacuum line.

CAUTION: IMPROPER SETTING WILL CAUSE DAMAGE TO CABIN PRESSURIZATION CONTROLLER.

(c) Turn rate selector on cabin pressurization controller full clockwise. Turn cabin altitude selector full counter-clockwise.

(d) Ensure that pilot's storm window, emergency exit, and cabin door are properly installed and secured closed. Strap or net cabin door.

NOTE: Perform the following steps outside aircraft by monitoring test cart operation and inspecting gauge through cockpit window.

(e) Connect instrument line connector port on test cart to the 1/4 inch bulkhead fitting on forward pressure bulkhead.

(f) Disconnect flexible connectors at bleed air supply connections on firewall. (See "Figure 1" on page 21202 and "Figure 1" on page 21404.)

(g) Connect test cart air outlet port to either left or right flexible connector. Plug opposite flexible connector.

WARNING: DO NOT OCCUPY CABIN DURING TESTS.

CAUTION: TEST CART OPERATOR MUST CONTINUALLY MONITOR STAND OPERATION DURING ALL TIMES THAT CABIN IS PRESSURIZED. DURING INITIAL PRESSURIZATION, ENSURE BY INSPECTION THROUGH COCKPIT WINDOW THAT AIRCRAFT AND CABIN ALTITUDE, AIRSPEED, AND RATE OF CLIMB INDICATORS ARE STABLE AND UNAFFECTED BY CABIN PRESSURE. ANY OF THE ABOVE INSTRUMENTS BEING AFFECTED INDICATES A LEAK IN THE STATIC SYSTEM. CORRECT BEFORE PROCEEDING.

(h) Start airflow into fuselage slowly and at a rate not to exceed 1,500 feet per minute rate of change in cabin altitude.

(i) When cabin differential gauge indicates one (1) pound per square inch differential (PSID), maintain only sufficient airflow to hold one (1) PSID. Inspect complete cabin structure, bleed air system, outflow and safety valves, etc. Identify major leaks for corrective sealing/repair.

(j) Continue to pressurize the cabin at a rate not to exceed 1,500 feet per minute until a maximum cabin differential of 5.25 PSID occurs. Determine airflow required to maintain 5.25 PSID and record. If airflow exceeds 40 SCFM*, or a major leak is occurring in one or more places, or if excessive noise is found, identify for corrective sealing. Some airflow leaking/exiting from the bottom of the cabin pressure safety and outflow valves is acceptable.

$$^{*}\ \text{SCFM} = \text{CFM obs}\left[\left(\sqrt{\frac{\text{P obs} + 29.92}{29.92}}\right)\left(\sqrt{\frac{520\ ^{\circ}\text{F}}{\text{T obs} + 460\ ^{\circ}\text{R}}}\right)\right]$$

Where: P obs = PSID observed x 2.036
T obs = Temperature (˚F) observed
CFM obs = Cubic Feet per Minute observed
SCFM = Standard Cubic Feet per Minute (flow)

NOTE: If cabin will not pressurize above 1 PSID due to excessive leakage:

(a) Disconnect cabin controller line at outflow valve.
(b) Pressurize to not more than 5.25 PSID to identify all leaks.
(c) Repair leaks.
(d) When an acceptable leak rate is achieved, depressurize cabin.
(e) Connect cabin controller line to outflow valve.

WARNING: DO NOT EXCEED 5.60 MAXIMUM DIFFERENTIAL PRESSURE IN NEXT STEP. EXCESSIVE PRESSURE CAN CAUSE STRUCTURAL DAMAGE OR PERSONAL INJURY.

(k) Continue to pressurize the cabin at a rate not to exceed 1,500 feet per minute until either the outflow or safety valve starts to relieve. Note differential pressure and which valve operated. Correct operating range is 5.50 + 0.1, - 0.20 PSID.
(l) Lower cabin pressure to 4.0 ± 0.2 PSID.
(m) Plug the ambient sensing port of the valve which operated above.

WARNING: DO NOT EXCEED 5.60 MAXIMUM DIFFERENTIAL PRESSURE IN NEXT STEP. EXCESSIVE PRESSURE CAN CAUSE STRUCTURAL DAMAGE OR PERSONAL INJURY.

(n) Slowly increase cabin pressure at a rate not to exceed 1,500 feet per minute and observe the differential pressure at which the remaining valve starts to relieve. Record. Correct operating range is 5.50 + 0.1, - 0.20 PSID.
(o) Lower cabin pressure to 4.0 ± 0.2 PSID.
(p) Unplug the ambient sensing port of the valve which operated first and was subsequently plugged.

WARNING: DO NOT EXCEED 5.60 MAXIMUM DIFFERENTIAL PRESSURE IN NEXT STEP. EXCESSIVE PRESSURE CAN CAUSE STRUCTURAL DAMAGE OR PERSONAL INJURY.

(q) Repeat step (k). Verify that the same valve operates first and that it operates within the correct pressure range (i.e. - 5.50 + 0.1, - 0.20 PSID).

(r) Depressurize cabin at a rate not exceeding 1,500 fpm.

(s) Remove instrument line connector and pressurization outlet connections installed in steps (e) thru (g). Plug 1/4 inch fitting in forward pressure bulkhead used for instrument line connector.Unstrap (or remove net from) cabin door.

(t) Uncap, and reconnect vacuum line and tee at controller.

(u) Turn rate and cabin altitude selectors on cabin controller full counterclockwise.

(v) Reconnect left and right flexible connectors at firewall to bleed air supply lines.

(w) Reinstall cowling.

B. Electro-Mechanical Pressurization System (S/N's 4636633, 4636652 and up) (PIR-PPS60144-5, Rev. NEW.)

(1) Equipment Required

Pressurization Test Cart (commercially available)

(2) Test Setup

(a) Connect test cart to the airplane as follows:

1) Connect instrument line connector port on test cart to 1/4 inch bulkhead fitting on the cabin forward pressure bulkhead.

2) Connect pressurization air outlet port to either left or right flexible connectors at bleed air system supply connections. Plug off opposite flexible connector.

(b) Verify aircraft status; Weight On Wheels (Aircraft not on jacks, LH main gear squat switch open).

(c) Verify all static air ports on the belly of the aircraft are unobstructed.

(d) Apply power to the aircraft.

(e) Verify Cabin Dump Switch is not activated.

(f) On LH AFT Circuit Breaker Panel verify that the CABIN PRESS CONTROL and CABIN PRESS DUMP circuit breakers are in.

WARNING: FAILING TO PULL THE HYD. PUMP POWER MAY RESULT IN ACCIDENTAL LANDING GEAR COLLAPSE.

(g) On the LH FWD Circuit Breaker Panel PULL the HYD. PUMP POWER Circuit breaker.

(h) Verify CPCS Controller (RH Side forward of RH Outflow Valve) is operating. The internal vacuum pumps should provide an audible indication that the controller is operating.

(i) Upon application of power in step (d), the CPCS controller will perform a BIT (Built In Test), wait three (3) minutes for the test to complete or until the vacuum pumps inside the controller stop.

(j) Press the TMR/REF button on the PFD, then set the landing field elevation (LFE) to -990 feet.

(k) Change aircraft status to Weight Off Wheels. (Close LH Main Gear Squat Switch).

(l) Wait two (2) minutes for CPCS controller to inflate Outflow Valve diaphragms fully against the valve seat. Do Not Exceed three (3) minutes, at three (3) minutes the controller automatically reverts to ground mode. If this occurs repeat from step (a) above.

(m) On the LH AFT Circuit Breaker Panel PULL the CABIN PRESS CONTROL and CABIN PRESS DUMP circuit breakers.

(n) Ensure that the emergency exit and entry door are properely installed and secured closed. Strap or net the cabin door.

(3) Test Procedure

WARNING: DO NOT OCCUPY CABIN DURING TESTS.

WARNING: DO NOT EXCEED 5.70 MAXIMUM DIFFERENTIAL PRESSURE IN NEXT STEP. EXCESSIVE PRESSURE CAN CAUSE STRUCTURAL DAMAGE OR PERSONAL INJURY.

CAUTION: TEST CART OPERATOR MUST CONTINUALLY MONITOR STAND OPERATION DURING ALL TIMES THAT CABIN IS PRESSURIZED. DURING INITIAL PRESSURIZATION, ENSURE BY INSPECTION THROUGH COCKPIT WINDOW THAT AIRCRAFT AND CABIN ALTITUDE, AIRSPEED, AND RATE OF CLIMB INDICATORS ARE STABLE AND UNAFFECTED BY CABIN PRESSURE. ANY OF THE ABOVE INSTRUMENTS BEING AFFECTED INDICATES A LEAK IN THE STATIC SYSTEM. CORRECT BEFORE PROCEEDING.

(a) Start airflow into fuselage slowly and a at rate not to exceed 1500 feet per minute (FPM) rate of change (Cabin Altitude).

(b) When the cabin differential gauge indicates one (1) pound per square inch differential (PSID), maintain only sufficient air flow to hold one (1) PSID and inspect complete cabin structure, bleed system, outflow valves, etc. for maor leaks and identify them for corrective sealing/repair action.

WARNING: DO NOT EXCEED 5.70 MAXIMUM DIFFERENTIAL PRESSURE IN NEXT STEP. EXCESSIVE PRESSURE CAN CAUSE STRUCTURAL DAMAGE OR PERSONAL INJURY.

(c) Continue to pressure the cabin at a rate not to exceed 1500 FPM until a maximum cabin differential of 5.25 +/- 0.1 PSID occurs.

(d) If the leakage rate meets/exceeds 40 SCFM, repair leaks then repeat test. If the leakage rate is below 40 SCFM continue to the next step.

(e) At the outflow valve static reference ports on the LH side belly of the aircraft, apply aluminum “Speed” tape to the forward static air port.

(f) Increase cabin pressure slowly at a reat not to exceed 1500 FPM. Verify that the outflow valves open to relieve cabin pressure between 5.4 and 5.7 PSID.

NOTE: 5.6 PSID is the maximum cabin delta P allowed, however , to acount for pressure losses between the measurement source and the outflow valve delta P limiter it is acceptable to increase the delta P indication at the cart to 5.7 PSID.

(g) Reduce cabin pressure slowly at a rate not to exceed 1500 FPM until the cabin is depressurized. Remove the aluminum tape from the forward static air port.

(h) Reset Test Setup (above) steps “(b)” thru “(n)” such that the outflow valves are both closed and doors are securely closed.

(i) At the outflow valve static reference ports on the LH side belly of the aircraft, apply aluminum “Speed” tape to the aft static air port.

(j) Increase cabin pressure slowly at a reat not to exceed 1500 FPM. Verify that the outflow valves open to relieve cabin pressure between 5.4 and 5.7 PSID.

(k) Reduce cabin pressure slowly at a rate not to exceed 1500 FPM until the cabin is depressurized. Remove the aluminum tape from the forward static air port.

(l) Remove test cart, by reversing the installation procedure. Plug 1/4 inch fitting in forward pressure bulkhead used for instrument line connector.Unstrap (or remove net from) cabin door.

(m) Unstrap or remove netting from cabin door and open.

(n) Return aircraft to Weight On Wheels status. Set LFE to zero (0) feet. Push in the CABIN PRESS CONTROL and CABIN PRESS DUMP circuit breakers. Check the landing gear switch is in the down position before pushing HYD. PUMP POWER circuit breaker.

(o) Remove power from aircraft.

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HEATING

1. Heat Exchanger/Air Selector Valve/Plenum Assembly (PA-46-350P Only)

   A. Removal

      (1) Loosen band clamp and remove flexible hose from bottom of fiberglass discharge plenum.

      (2) Loosen band clamps and remove flexible hose from bleed air supply manifold (part of heat exchanger/selector valve assembly).

      (3) Disconnect and remove cabin pressurization control cable and bleed air dump cable from air selector valve/heat exchanger assembly.

      (4) Remove the four bolts and washers that secure heat exchanger/selector valve assembly brackets to mounting brackets on baggage compartment skin.

      (5) Remove six bolts, nuts, and washers from the two inch diameter bleed air tube at the firewall.

      (6) Remove the four bolts attaching air temperature control valve to forward side of firewall.

      (7) Remove assembly from aircraft.

   B. Installation

      (1) Align assembly in proper position as shown in “Figure 1” on page 21404.

      (2) Attach air temperature control valve to forward side of firewall with four bolts.

      (3) Attach six bolts and hardware to the two inch diameter bleed air tube at the firewall.

      (4) Secure heat exchanger/selector valve assembly brackets to mounting brackets on baggage compartment skin with four bolts and washers.

      (5) Connect cabin pressure control cable and bleed air dump cable to air selector valve/heat exchanger assembly.

      (6) Attach flexible hose to bleed air supply manifold (part of heat exchanger/selector valve assembly) and tighten band clamps.

      (7) Attach flexible hose to bottom of fiberglass discharge plenum and tighten band clamp.

2. Air Selector Valve (PA-46-350P Only)

   NOTE: Remove heat exchanger/air selector valve/plenum assembly before individually removing air selector valve.

   A. Removal

      (1) Loosen closure assembly (arm) and slide it to the right, off of the selector crank assembly.

      (2) Remove the four bolts that attach the check valve assembly (including gaskets) to the air selector valve.

      (3) Remove the twelve screws attaching the air selector valve to the heat exchanger.

      (4) Remove selector valve from heat exchanger.

   B. Installation

      (1) Position air selector valve onto heat exchanger.

      (2) Attach air selector valve to heat exchanger with twelve screws.

      (3) Attach check valve assembly (including gaskets) to air selector valve with four bolts.

      (4) Slide closure assembly (arm) onto the selector crank assembly and tighten.

3. Heat Exchanger (PA-46-350P Only)

NOTE: Remove heat exchanger/air selector valve/plenum assembly before individually removing heat exchanger.

A. Removal

(1) Remove four bolts and hardware attaching bleed air shutoff valve to heat exchanger.
(2) Remove twelve screws attaching heat exchanger to selector valve.
(3) Remove twelve screws attaching heat exchanger to plenum.
(4) Remove heat exchanger from assembly.

B. Installation

(1) Align heat exchanger into proper position.
(2) Attach heat exchanger to plenum with twelve screws.
(3) Attach heat exchanger to selector valve with twelve screws.
(4) Attach bleed air shutoff valve to heat exchanger with four bolts and hardware.

4. Plenum (PA-46-350P Only)

NOTE: Remove heat exchanger/air selector valve/plenum assembly before individually removing plenum.

A. Removal

(1) Remove twelve screws attaching plenum to heat exchanger.
(2) Remove plenum.

B. Installation

(1) Seal plenum to heat exchanger with RTV 106 sealant.
(2) Install plenum to heat exchanger with twelve screws.

5. Bleed Air Shutoff Valve (PA-46-350P Only)

A. Removal

(1) Disconnect control linkage to shutoff valve.
(2) Loosen bleed air flexible hose band clamps.
(3) Remove four bolts and hardware attaching shutoff valve (including gaskets) to heat exchanger.
(4) Remove valve.

B. Installation of Bleed Air Shutoff Valve

(1) Align shutoff valve into proper position.
(2) Attach shutoff valve (including gaskets) to heat exchanger with four bolts and hardware.
(3) Tighten bleed air flexible hose band clamps.
(4) Connect control linkage to shutoff valve.
(5) Check rigging of shutoff valve.

6. Auxiliary Cabin Heater

See “Figure 1” on page 21404.

A. Removal

(1) Loosen clamps attaching flex adapter to aft side of blower.
(2) Slide adapter forward over blower.
(3) Remove the six bolts attaching cabin air supply tube and auxiliary heater to forward pressure bulkhead.
(4) Disconnect electrical leads to heater.
(5) Remove heater.

B. Installation

(1) Apply GE RTV 1508 sealant to gasket.
(2) Position heater and gasket on forward side of bleed air check valve at forward pressure bulkhead.
(3) Secure heater, check valve, and cabin air supply tube to forward pressure bulkhead with six bolts.
(4) Slide flexible blower adapter duct aft over heater and secure with band clamps.
(5) Connect electrical leads.

7. Defroster

See “Figure 1” on page 21404.

A. Removal

(1) Remove closeout panel covering magnetic (float) compass and windshield defroster by removing compass deviation and removing screw securing deviation card holder and close out panel
(2) Remove the four screws and nylon spacers securing defroster to windshield post.
(3) Loosen clamps securing defrost hoses to defroster. Remove hoses.
(4) Remove defroster.

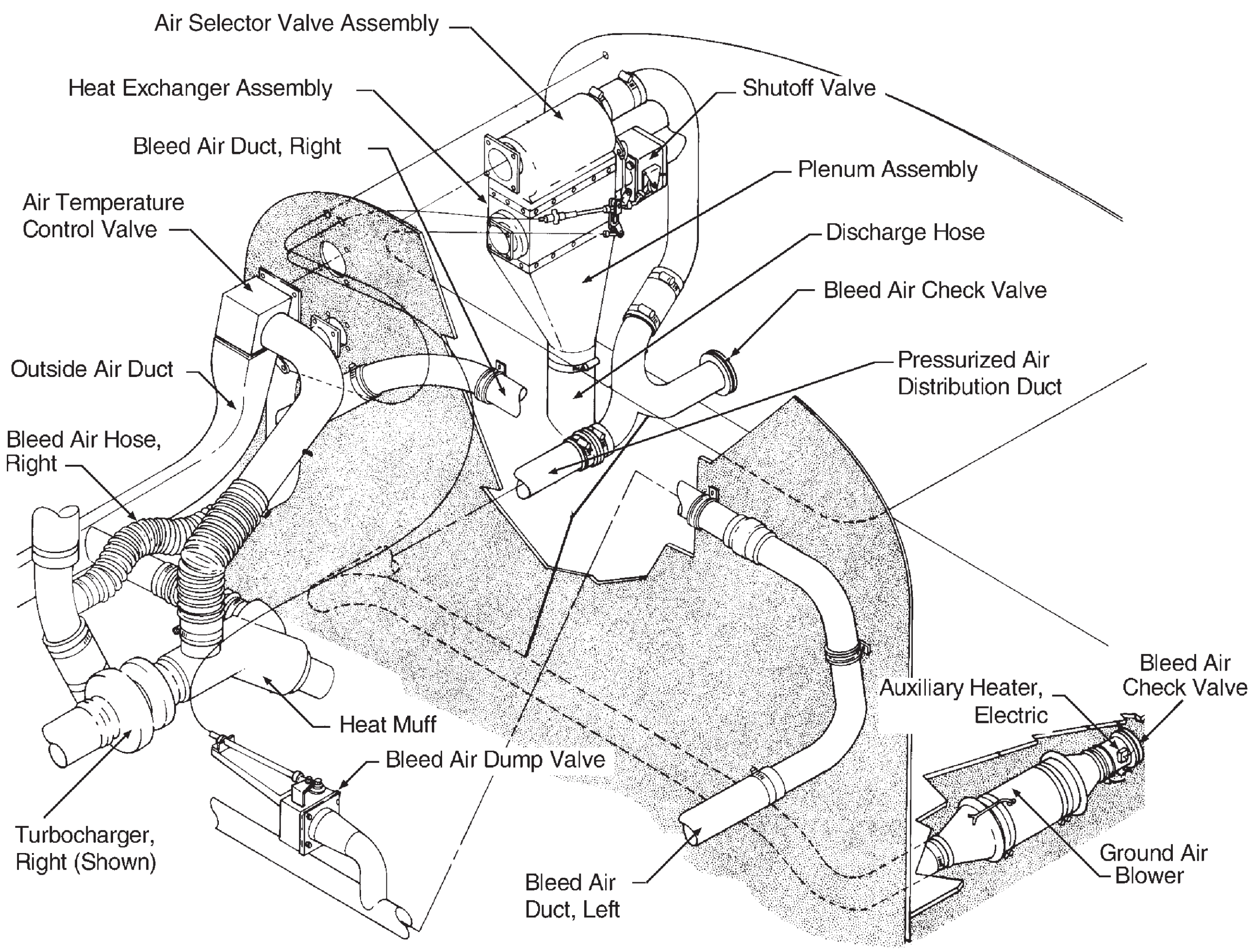
B. Installation

(1) Connect defrost hoses to appropriate fittings on defroster. Tighten clamps screws.
(2) Position defroster at windshield post. Secure with four nylon spacers and screws
(3) Position closeout panel in appropriate position over defroster and magnetic (float) compass
(4) Position compass deviation card holder in proper location. Be sure screw hole in deviation card holder aligns with screw hole in closeout panel. Secure both in position with brass screw previously removed.

NOTE: Only a brass screw may be used to secure compass card holder and close out panel to windshield post.

C. Rigging

(1) Push defroster control knob in.
(2) Ensure that the defroster restrictor valve is fully open.



Effectivity
4636001 and Up

Cabin Heat, Defroster, and Fresh Air System
Figure 1 (Sheet 1 of 5)

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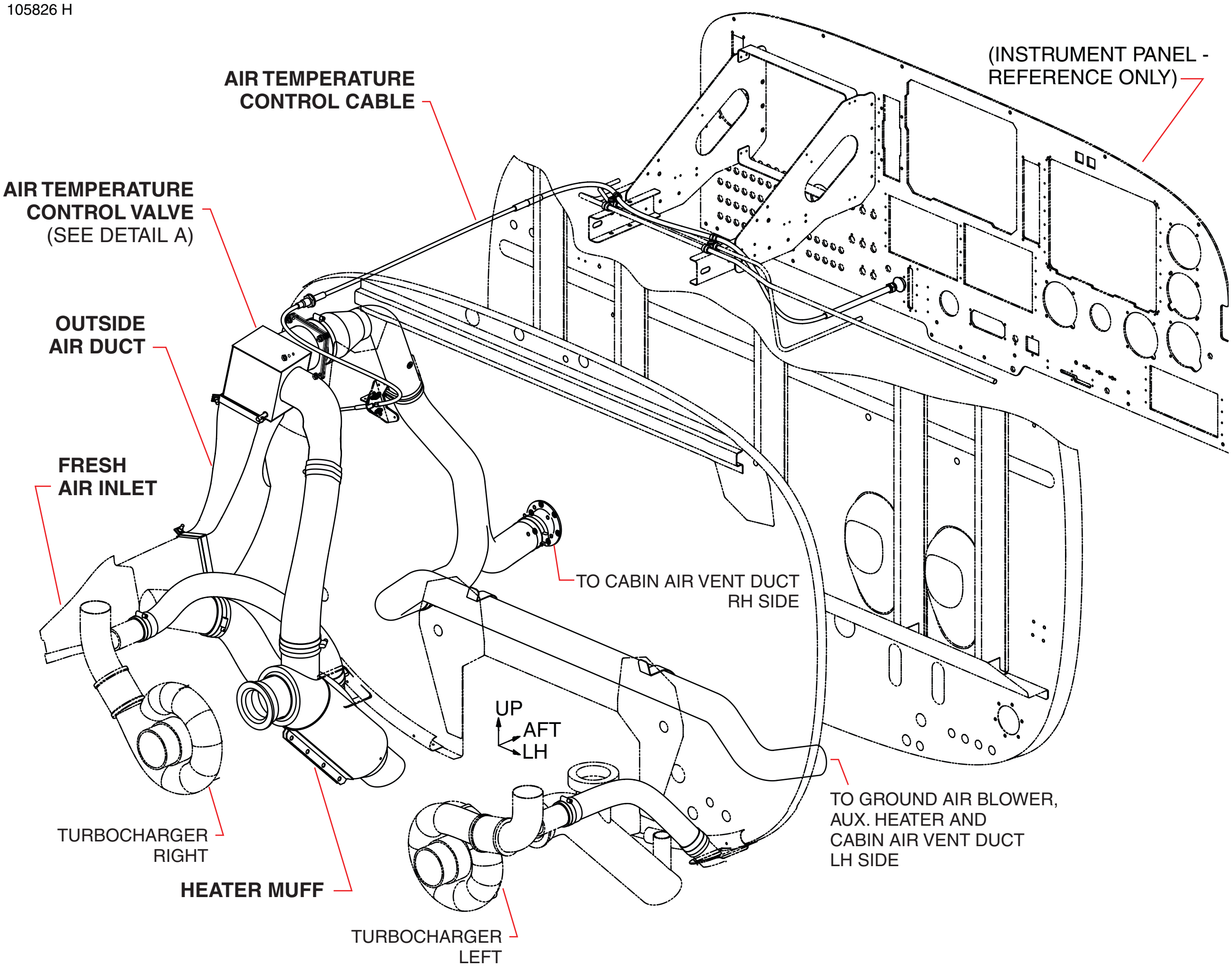
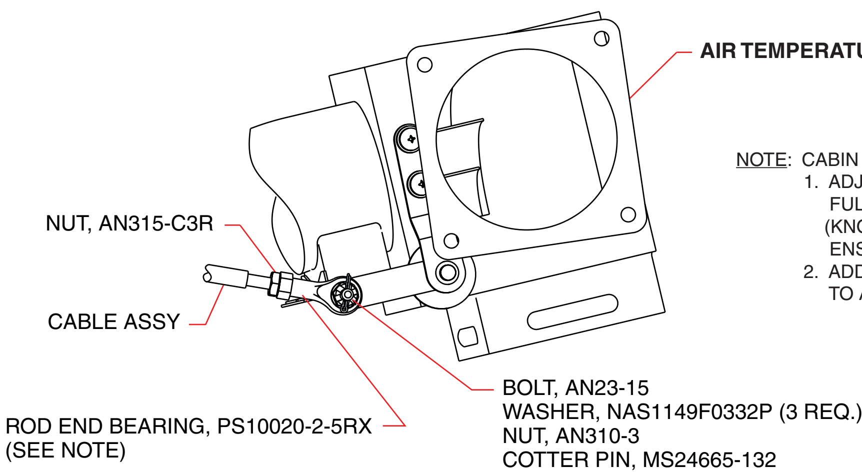


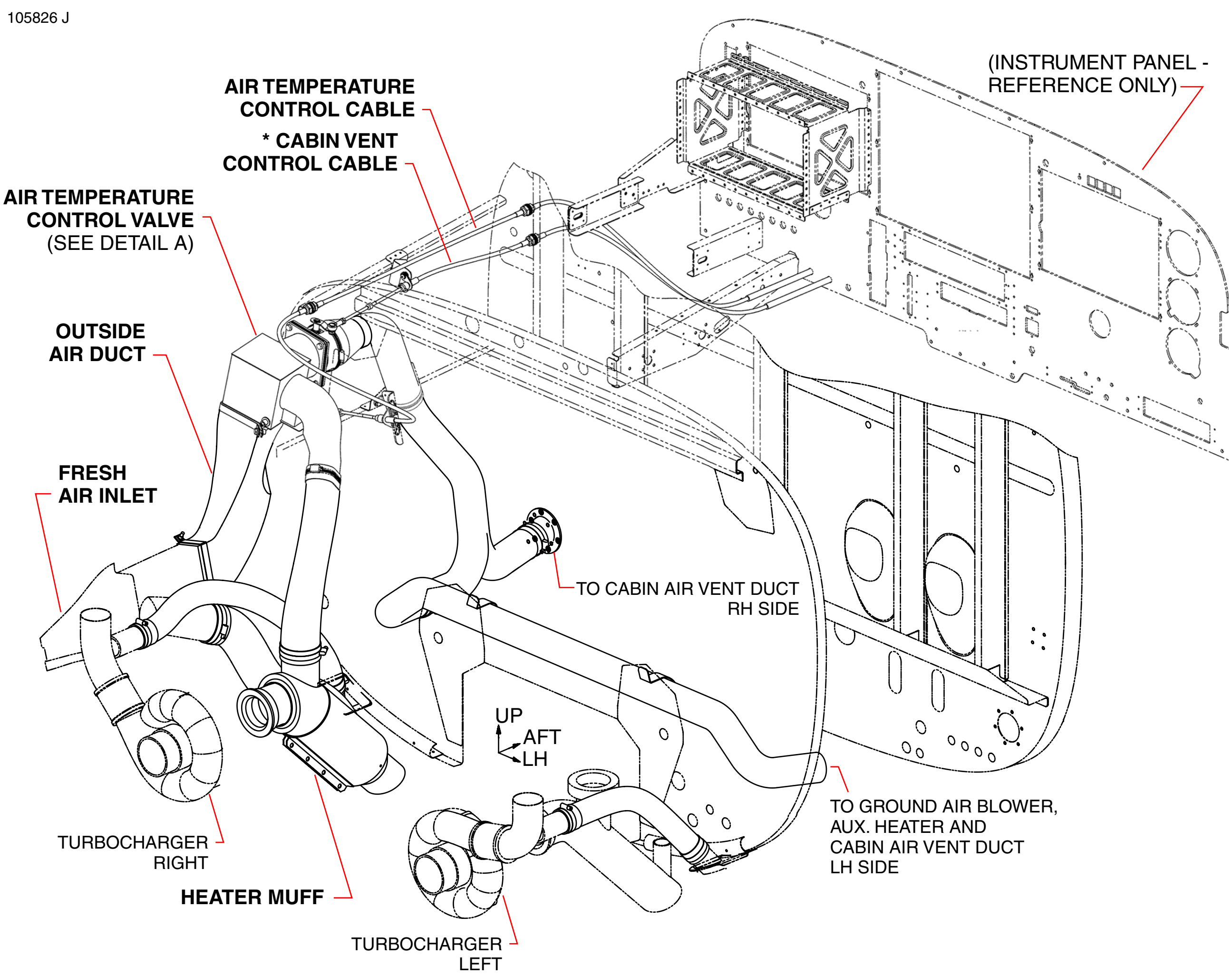


NOTE: CABIN TEMPERATURE RIGGING
1. ADJUST ROD END BEARING IN OR OUT TO ALLOW FULL CLOSURE OF HEAT MUFF HOT AIR FLOW (KNOB IS PUSHED INTO INSTRUMENT PANEL) AND ENSURE DUMP DOOR IS OPEN FULLY.
2. ADD OR REMOVE ONE SPACER, IF REQUIRED, TO ACHIEVE ADJUSTMENT.

DETAIL A
(LOOKING FORWARD)

Cabin Heat, Defroster, and Fresh Air System
Figure 1 (Sheet 2 of 5)

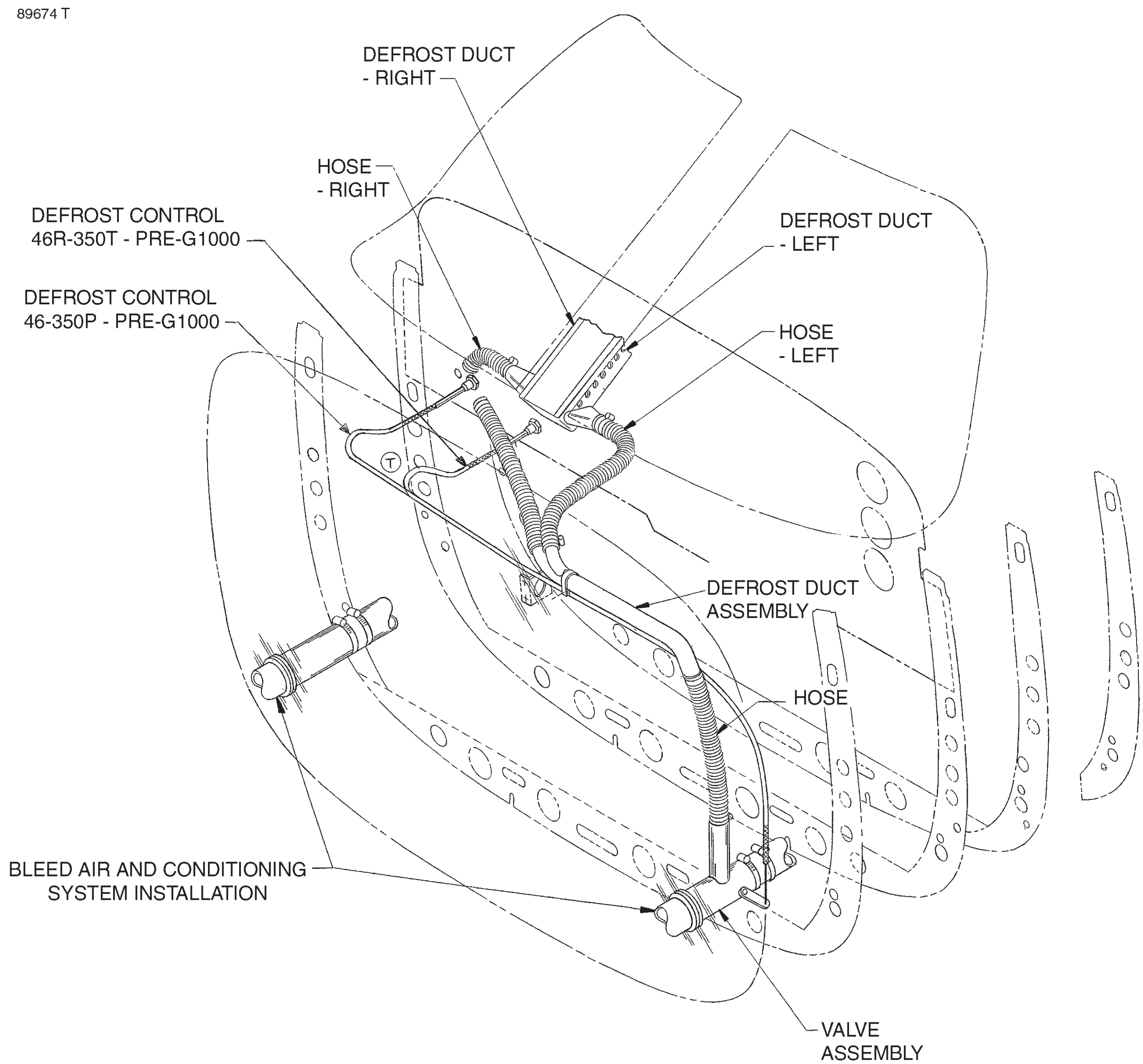
Effectivity
4692001 and Up



* CABIN VENT CONTROL - S/N 4692165 AND UP

Effectivity
4692134 and Up with G1000

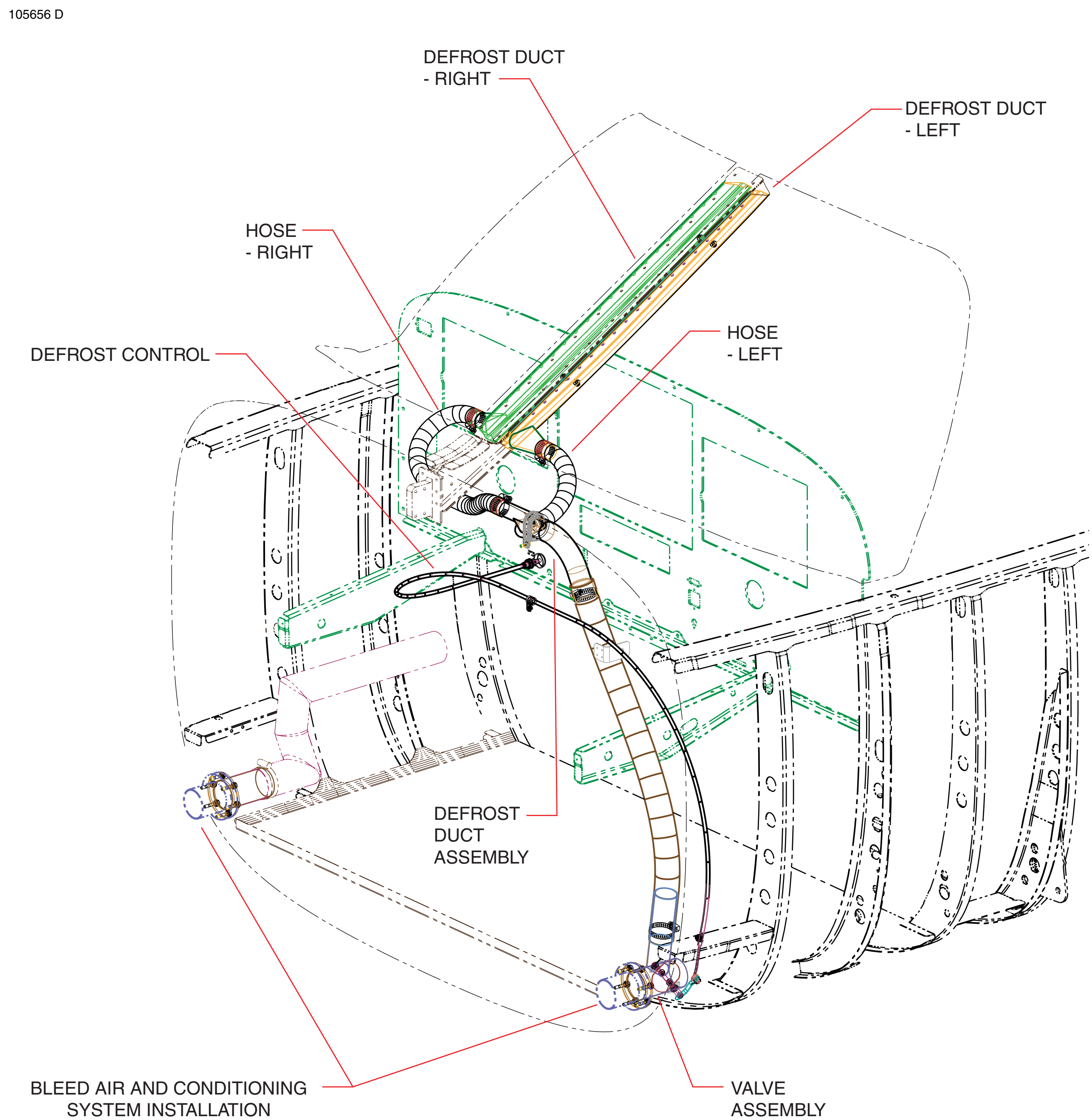
Cabin Heat, Defroster, and Fresh Air System
Figure 1 (Sheet 3 of 5)



DEFROST - DUCT AND CABLE LOCATION

Cabin Heat, Defroster, and Fresh Air System
Figure 1 (Sheet 4 of 5)

Effectivity
4636001 and Up
4692001 and Up



DEFROST - DUCT AND CABLE LOCATION

Effectivity
4636460,
4636463 and Up with G1000
4692134 and Up with G1000

Cabin Heat, Defroster, and Fresh Air System
Figure 1 (Sheet 5 of 5)

COOLING

1. General

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

WARNING: REFRIGERANT R12 IS USED IN S/N'S 4636001 THRU 4636186 ONLY. REFRIGERANT HFC-134A IS USED IN S/N'S 4636187 AND UP AND S/N'S 4692001 AND UP. ANYONE SERVICING THE AIR CONDITIONING SYSTEM MUST BE FAMILIAR WITH THE REFRIGERANT, LUBRICANT, AND COMPONENTS USED IN THAT PARTICULAR SYSTEM.

NOTE: The air conditioning system should be operated at least once a month to prevent sticking valves and to keep the system lubricated.

NOTE: "HFC-134a" means air conditioning refrigerant which is generically identified and documented as 1,1,1,2-Tetrafluoroethane and/or CAS# 811-97-2. Brand names and commercial designations which meet these requirements include (but are not limited to):

HFC134a, HFA134a, R134a, Dymel® 134a, Fluorocarbon 134a, Forane® 134a, Genetron® 134a, Halocarbon 134a, KLEA® 134a, Norfluane, Referon® 134a, and SUVA™ 134a.

This installation consists of a compressor with its special brackets, two evaporator modules and blowers, a condenser, a receiver-dehydrator, a manifold assembly which incorporates dual pressure switches, high and low service ports, and related plumbing.

The evaporators filter, dehumidify and cool the air. The evaporator modules are mounted underneath the floor between the ribs at F.S. 233.87 and F.S. 241.50 along with the blowers, expansion valves, ducting, service ports, and related tubing.

The compressor is a five cylinder, rotary, piston type which is supported by special brackets at the forward, left area of the engine. A V-belt connected to the starter ring gear pulley, drives the compressor through a magnetic clutch.

The condenser is located in the aft fuselage. Air is drawn through the condenser and vented overboard by means of a blower and duct.

The manifold assembly incorporates high and low service ports, a high pressure switch, and a low pressure switch that control the condenser head pressure by engaging or disengaging the compressor via a magnetic clutch.

The air conditioning system uses refrigerant HFC-134a, except in S/N's 4636001 thru 4636186 only - which use R-12. The refrigerant enters the compressor as a vapor. The compressor pressurizes the heat-laden vapor until its pressure and heat reach a point much hotter than the outside air. The compressor then pumps the vapor to the condenser where it cools and changes to a liquid. The liquid then passes to the receiver-dehydrator. The receiver-dehydrator's function is to filter, remove any moisture and ensure a steady flow of liquid refrigerant into the evaporators through the expansion valves. The expansion valves are temperature controlled metering valves which regulate the flow of liquid refrigerant to the evaporators. Inside each evaporator the liquid refrigerant changes state to a gas and, in doing so, absorbs heat. The evaporators then absorb the heat from the air passing over the coils. From the evaporators, the refrigerant vapor returns to the compressor where the cycle is repeated.

2. Troubleshooting

See "Chart 1" on page 21502.

CHART 1 (Sheet 1 of 4)
TROUBLESHOOTING (AIR CONDITIONING SYSTEM)

Trouble	Cause	Remedy
High discharge pressure.	Overcharge of refrigerant. Air in system.	Purge excess refrigerant. Check for leaks. Bleed charge from system. Evacuate and recharge system.
	Overheated condenser due to blocking air passage.	Clean bugs and dirt from condenser fins. Straighten fins if bent.
	Flooded evaporator indicated by heavy frosting on suction line and compressor suction service valve.	Replace expansion valve.
	Restriction in liquid line from condenser.	Check for kinked hoses and stopped up filter.
Low discharge pressure.	Undercharge of refrigerant. Sight glass shows bubbles or foam.	(R12) Add refrigerant until bubbles disappear. Check system for leaks.
		(HFC-134a) Check system for leaks. Evacuate and recharge system.
	Damaged compressor valves or dirt under valves.	Replace compressor.
	Damaged compressor. Worn or broken piston or piston rings.	Replace compressor.
Low suction pressure (Accompanied by icing evaporator.)	Low air supply through evaporator.	Repair blower or blower motor. Clean stoppage in air ducts.
	Very dirty evaporator fins and coils.	Clean and flush with water.
Low suction pressure. (Evaporator not cold enough.) Suction gauge may read a vacuum indicating evaporator lacks refrigerant.	Undercharge of refrigerant. Moisture freezing in expansion valve. Valve will show frost. Expansion valve inlet screen clogged. Inoperative expansion valve. Valve stuck closed.	Add refrigerant. Install new dryer. Evacuate and recharge. Remove screen. Clean with solvent and replace. If suction pressure does not change, replace expansion valve.
	Restriction anywhere in liquid line. Restriction will show frost.	Locate restriction and repair.

CHART 1 (Sheet 2 of 4)
TROUBLESHOOTING (AIR CONDITIONING SYSTEM)

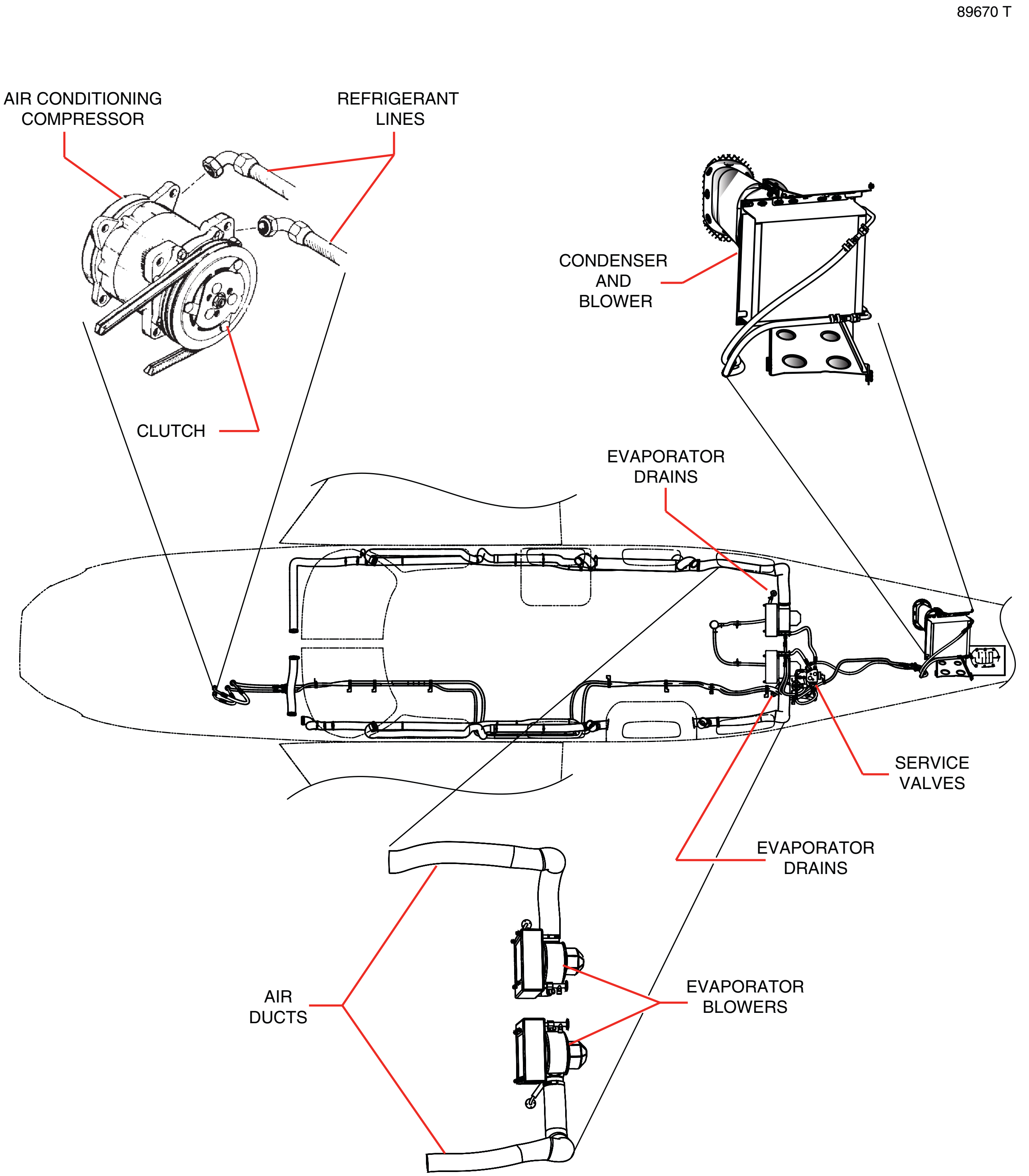
Trouble	Cause	Remedy
High suction pressure.	Expansion valve not closing. Evaporator flooded. Suction line frosted to compressor.	Replace expansion valve.
	Compressor drive belt slipping.	Adjust belt tension.
	Magnetic clutch slipping.	Check electrical circuit for correct voltage to clutch coil. Clean clutch surfaces of oil.
	Leaking or broken compressor valves.	Replace compressor.
System produces no cooling.	**Electrical**	
	Blown fuse in control head.	Replace fuse.
	Open circuit breaker.	Reset circuit breaker.
	Broken or disconnected electrical wire.	Check all terminals for loose connections; check wiring for hidden breaks.
	Broken or disconnected ground wire.	Check ground wire to see if loose, broken, or disconnected.
	Clutch coil burned out or disconnected.	Check current flow to clutch, replace if inoperative.
	Thermostat sensing element defective.	Check thermostat and cabin comfort control panel.
	Blower motor disconnected or burned out.	Check current flow to blower motor. Repair or replace if inoperative.
	Mechanical	
	Loose or broken drive belt.	Replace drive belts and/or tighten to specifications.
	Compressor partially or completely frozen.	Remove compressor for service or replacement.
	Expansion valve stuck in open position.	Replace expansion valve.

CHART 1 (Sheet 3 of 4)
TROUBLESHOOTING (AIR CONDITIONING SYSTEM)

Trouble	Cause	Remedy
System produces no cooling. (continued)	**Refrigeration**	
	Broken refrigerant line.	Examine all lines for evidence of breakage by external stress or rubbing wear.
	Leak in system.	Evacuate system, apply static charge, leak test system, and repair leak as necessary.
	Compressor shaft seal leaking.	Replace compressor.
	Clogged screen or screens in receiver-dehydrator or expansion valve; plugged hose or coil.	Repair as necessary.
System will not produce sufficient cooling.	**Electrical**	
	Blower motor sluggish in operation.	Remove blower motor for service or replacement.
	Mechanical	
	Compressor clutch slipping.	Remove clutch assembly for service or replacement.
	Obstructed blower passage.	Examine entire passage for obstruction. Correct as necessary.
	Insufficient air circulation over condenser coils; fins clogged with dirt or bugs.	Clean condenser coils.
	Evaporator filter clogged.	Clean with cleaning solvent to remove cigarette tars.
	Refrigeration	
	Insufficient refrigerant in system.	(R12) Recharge system until bubbles disappear in receiver-dehydrator and gauge readings stabilize to specifications.
		(HFC-134a) Recharge system until a minimum 20° F temperature drop is achieved at the evaporators during the Post Charging Operational Check and gauge readings stabilize to specifications.

CHART 1 (Sheet 4 of 4)
TROUBLESHOOTING (AIR CONDITIONING SYSTEM)

Trouble	Cause	Remedy
System will not produce sufficient cooling. (continued)	**Refrigeration** (continued)	
	Clogged screen in expansion valve.	Purge system and replace expansion valve.
	Clogged screen in receiver-dehydrator.	Purge system; replace receiver-dehydrator.
	Excessive moisture in system.	Purge system; replace receiver-dehydrator.
	Air in system.	Purge, evacuate and charge system. (Replace receiver-dehydrator)
Excessively noisy system.	**Mechanical**	
	Loose or excessively worn drive belt.	Tighten or replace as required.
	Noisy clutch.	Remove clutch for service or replacement as necessary.
	Compressor noisy.	Check mountings and repair; remove compressor for service or replacement.
	Compressor oil level low.	Fill with correct amount of specified oil.
	Electrical	
	Defective winding or improper connection in compressor clutch coil.	Replace or repair as necessary.
	Refrigeration	
	Excessive charge in system.	Discharge excess refrigerant until high pressure gauge drops within specifications.
	Low charge in system.	Check system for leaks; charge system.
	Excessive moisture in system.	Replace receiver-dehydrator; purge, evacuate, and charge system.



Air Conditioning System
Figure 1

3. Servicing Cooling System

(PIR-PPS50003-4, Rev. K)

NOTE: The maximum refrigerant capacity is 2.75 pounds. The total refrigerant capacity required is determined separately for each system and is the amount that will result in: bubble-free operation at the sight gauge (R12); or a minimum 15° F temperature drop at the evaporators (HFC-134a); as specified in the Post Charging Operational Check.

A. Definitions

(1) High Side. The "high side" consists of all lines and components between the compressor outlet and the expansion valves. It includes the condenser and receiver-dehydrator sight gage.

(2) Low Side. The "low side" consists of all lines and components between the expansion valves and the compressor inlet. It includes the evaporators.

B. Service Valves (i.e. - ports)

See "Figure 1" and "Figure 2" on page 21508.

(1) Description

The service valves are installed to Test, Bleed (Discharge), Evacuate and Charge the air conditioning system. This airplane is equipped with inline service valves mounted on the manifold assembly. All normal air conditioning service should be performed at the manifold assembly mounted valves.

NOTE: Service valves are also located on the compressor in S/N's 4636001 thru 4636186 only. However, use of these valves in servicing is not recommended.

NOTE: In S/N's 4636001 thru 4636186 only, the service valves are screw-on type Schrader valves.

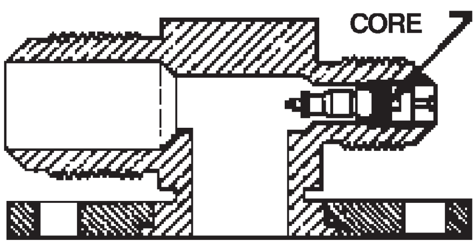
In S/N's 4636187 and up and S/N's 4692001 and up, the service valves are quick-disconnect type Schrader valves.

NOTE: If a Schrader service valve is not serviceable, the core assembly must be replaced.

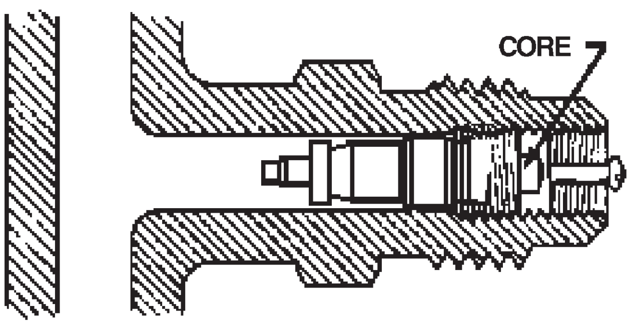
(2) Replacement

CAUTION: WHENEVER THE AIR CONDITIONING REFRIGERANT LINES OR SYSTEM ARE OPENED FOR ANY REASON, THE LINES AND FITTINGS SHOULD BE CAPPED AND SEALED IMMEDIATELY TO PREVENT DIRT AND OTHER CONTAMINANTS FROM ENTERING THE SYSTEM. (DO NOT PUT A PLUG INTO THE HOSES OR FITTINGS.)

When installing the valves on the manifold assembly, lubricate the threads with Tite Seal 3 (Do not apply to the first two threads).

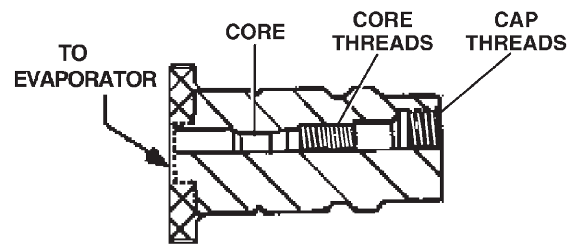


COMPRESSOR VALVE
(NOT RECOMMENDED FOR NORMAL SERVICING)

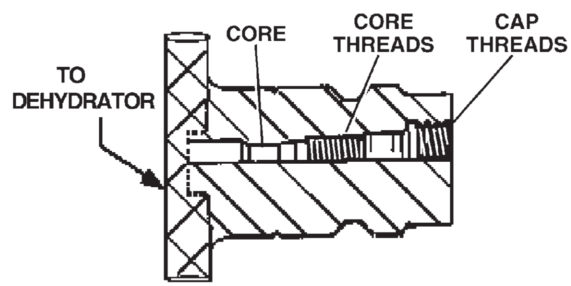


INLINE SERVICE VALVE

(S/N'S 4636001 THRU 4636186 ONLY)



LOW SIDE (PRESSURE) VALVE



HIGH SIDE (PRESSURE) VALVE

(S/N'S 4636187 AND UP)

Service Valves
Figure 2

21508

C. Malfunction Detection

NOTE: If the cooling system has leaked refrigerant or is discharged, the compressor oil level must be checked.

The detection of system malfunction largely depends on the mechanic's ability to interpret the gauge pressure readings into system problems. A system operating normally will have a low side gauge pressure reading that will correspond with the temperature of the refrigerant evaporating in the evaporator, allowing for a few degrees temperature rise due to loss in the tube walls and fins. The high side will have a gauge pressure that will correspond with the temperature of the refrigerant condensing in the condenser, allowing for a few degrees temperature drop due to loss in the tube walls and fins.

Any deviation from that which is normal indicates a malfunction within the system due to a faulty control device, obstruction, defective part, or improper installation.

Detection of system malfunction is made easier with knowledge of the relationship between temperature and pressure of the refrigerant (R-12 or HFC-134a). They are in close proximity between the pressures of twenty and eighty pounds per square inch (psi). A glance at the temperature-pressure chart (i.e. - "Chart 2" on page 215010) will show that there is only a slight variation between the temperature and pressure of the refrigerant in the lower range.

It is correct to assume that for every pound of pressure added to the low side, a temperature increase of about one degree Fahrenheit takes place. For instance, a pressure of 23.8 on the chart indicates a temperature of 24°F. A change of pressure of almost one pound to 24.6 psi gives us a temperature increase to 25°F.

NOTE: For each 1,000 feet of elevation above sea level, the gauge readings will be about one inch of mercury or 1/2 psi higher than the chart indicates.

It must be pointed out that the actual temperature of the air passing over the coils of the evaporator will be several degrees warmer than the fins, allowing for a temperature rise caused by the loss in the fins and tubing of the evaporator.

The importance of a seasonal check up of the air conditioning system should be brought to the attention of the customer whenever possible. A thorough check of the system performed in a methodical manner will reveal trouble the customer is often not aware of. Locating and repairing the trouble early will usually result in savings to the customer both in time and additional troubles that too often result from neglect.

A performance test of the system is the only positive way in which the complete system can be checked for efficient operation. The air conditioning system should be given this test before work is begun on the system whenever possible. However, if the system is completely inoperative, repairs must be performed before the system can be properly tested. The test can uncover further work that must be performed before the system is brought to its full operating efficiency. The performance test should always be performed after repair work has been done and before the airplane is released to the customer. The serviceman performing this test carefully will ensure that the repairs have been properly performed and that the system will operate satisfactorily.

NOTE: The air conditioning system should be operated at least once a month to prevent sticking valves and to keep the system lubricated.

The performance test, when properly performed includes a thorough examination of the outside of the system as well as the inside. Many related parts are overlooked because it is felt they are of no bearing on the operating efficiency of the unit. For this reason, a thorough visual inspection of the complete system should be performed, followed by an operating inspection of the system.

CHART 2
TEMPERATURE VS. PRESSURE

Refrigerant R12 Evaporator Pressure Gauge Reading psi	Refrigerant R12 Evaporator Temperature °F	Refrigerant HFC134a Evaporator Pressure Gauge Reading psi	Refrigerant HFC134a Evaporator Temperature °F
0	-21	-5	-27
2.4	-15	0	-15
4.5	-10	2	-9
10.1	2	4	-4
11.2	4	6	0
12.3	6	8	4
13.4	8	10	7
14.6	10	12	11
15.8	12	14	14
17.1	14	16	17
18.3	16	18	20
19.7	18	20	22
21	20	22	25
22.4	22	24	28
23.1	23	26	30
23.8	24	28	33
24.6	25	30	35
25.3	26	32	37
26.1	27	34	39
26.8	28	36	41
27.6	29	38	43
28.4	30	40	45
29.2	31	42	47
30	32	44	49
30.9	33	46	51
31.7	34	48	53
32.5	35	50	54
33.4	36	55	58
34.3	37	60	62
35.1	38	65	66
36	39	70	69
36.9	40		
37.9	41		
38.8	42		
39.7	43		
41.7	45		
43.6	47		
45.6	49		
48.7	52		
49.8	53		
55.4	57		
60	62		
64.9	66		

D. Special Servicing Procedures

The air conditioning system should be serviced by a qualified shop with trained personnel. The following procedures and precautions should be observed.

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS REQUIRE THAT AIR CONDITIONING SYSTEM REPAIRS BE ACCOMPLISHED BY A QUALIFIED SHOP WITH APPROPRIATELY TRAINED PERSONNEL.

The efficiency of this system depends upon the pressure-temperature relationship of pure refrigerant. As long as the system contains only pure refrigerant plus a specified amount of compressor oil (which is mixed with the refrigerant), it is considered to be chemically stable. Foreign materials within the system will affect the chemical stability, contaminate the system, and decrease its efficiency.

(1) Refrigerant Safety Precautions.

WARNING: REFRIGERANTS R12 (“FREON”) AND HFC-134A ARE ODORLESS AND COLORLESS IN EITHER THE LIQUID OR GASEOUS STATE. REFRIGERANT FOR CHARGING REFRIGERATION SYSTEMS IS SUPPLIED IN PRESSURIZED CONTAINERS (APPROX. 70 PSI AT 70°F) IN LIQUID FORM. SINCE THIS MATERIAL IS ESSENTIALLY INERT AT ROOM TEMPERATURES THE DANGERS ARE PRIMARILY ASSOCIATED WITH THE PRESSURE AND THE REFRIGERATION EFFECTS OF THE RELEASE AND SUBSEQUENT EVAPORATION OF THIS PRESSURIZED LIQUID.

WARNING: WEAR SUITABLE EYE PROTECTION WHEN HANDLING REFRIGERANT DUE TO THE POSSIBILITY OF FREEZING OF THE EYE IF CONTACTED BY ESCAPING LIQUID REFRIGERANT. IF LIQUID REFRIGERANT DOES STRIKE THE EYE, THE FOLLOWING ACTIONS SHOULD BE TAKEN:

(1) DO NOT RUB THE EYE.
(2) SPLASH LARGE QUANTITIES OF COOL WATER INTO THE EYE TO RAISE THE TEMPERATURE.
(3) TAPE ON AN EYE PATCH TO AVOID THE POSSIBILITY OF DIRT ENTERING THE EYE.
(4) RUSH TO A PHYSICIAN OR HOSPITAL FOR IMMEDIATE PROFESSIONAL AID.
(5) DO NOT ATTEMPT TO TREAT IT YOURSELF.

WARNING: WEAR SUITABLE CLOTHING AND GLOVES. IF LIQUID REFRIGERANT STRIKES THE SKIN, FROSTBITE CAN OCCUR. IF EXPOSED, TREAT WITH WARM (NOT HOT) WATER AND SEEK MEDICAL ATTENTION.

WARNING: DO NOT DISCHARGE LARGE QUANTITIES OF REFRIGERANT INTO CLOSED ROOMS. IT MAY DISPLACE MOST OF THE AIR IN THE ROOM AND THIS COULD CAUSE OXYGEN STARVATION. GASEOUS REFRIGERANT IS HEAVIER THAN AIR AND FLOWS TO THE BOTTOM OF A CONTAINER.

WARNING: DO NOT DISCHARGE REFRIGERANT INTO AN OPEN FLAME OR ONTO A VERY HOT SURFACE (500°F). POISONOUS PHOSGENE GAS IS GENERATED BY THE ACTION OF THE HEAT ON REFRIGERANT R12.

WARNING: DO NOT APPLY DIRECT FLAME OR OTHER HIGH HEAT SOURCE TO A REFRIGERANT CONTAINER DUE TO THE HIGH PRESSURES WHICH WILL RESULT. IF ANY HEATING IS DONE TO REFRIGERANT CONTAINERS THE CONTAINER PRESSURE SHOULD BE MONITORED AND KEPT BELOW 150 PSI.

(2) System Servicing Precautions.

WARNING: DISCHARGE SYSTEMS SLOWLY TO PREVENT THE ESCAPE OF LIQUID REFRIGERANT AND THE LOSS OF THE LUBRICATING OIL.

NOTE: The term "Discharge," as used throughout this section, in no sense implies or suggests discharging refrigerant to atmosphere. In all cases when discharging, an environmentally approved refrigerant recovery station is to be used.

WARNING: DO NOT LEAVE SYSTEMS OPEN TO THE ATMOSPHERE WHEN DISCHARGED. MOISTURE AND OTHER CONTAMINATION MAY ENTER AND DAMAGE OPEN SYSTEMS.

NOTE: A very strong acid (HCL) is formed when R-12 comes in contact with moisture. When HFC-134a comes in contact with moisture it absorbs it into the system, which will lead to system failure.

WARNING: USE ONLY APPROVED REFRIGERATION OIL IN THE COMPRESSOR:

-- MINERAL OIL FOR R12 SYSTEMS (S/N'S 4636001 THRU 4636186 ONLY); AND,

-- FOR HFC-134a SYSTEMS (S/N'S 4636187 AND UP AND S/N's 4692001 AND UP) USE POLYALKYLENE GLYCOL (PAG) OR POLYOL ESTER (POE)

IF ANY DOUBT EXISTS ABOUT THE CLEANLINESS OF THE COMPRESSOR OIL, REPLACE IT WITH NEW OIL.

WARNING: NEVER INTRODUCE ANYTHING BUT PURE REFRIGERANT AND THE APPROPRIATE REFRIGERANT OIL INTO A SYSTEM.

WARNING: KEEP REFRIGERANT OIL CONTAINERS TIGHTLY SEALED AND CLEAN TO PREVENT ABSORPTION OF MOISTURE OR OTHER CONTAMINATION.

WARNING: NEVER REUSE OIL REMOVED FROM THE SYSTEM -- DISCARD IT.

CAUTION: WHEN LOCTITE REFRIGERANT SEALANT HAS BEEN USED ON A JOINT IT MUST BE HEATED TO 400°F PRIOR TO DISASSEMBLY. LOCTITE MUST BE USED TO SEAL ANY PIPE THREADS IN THE SYSTEM LINES.

CAUTION: REPLACE THE RECEIVER-DEHYDRATOR ASSEMBLY ON ANY SYSTEM WHICH HAS BEEN OPERATING WITH A LEAK ALLOWING AIR TO ENTER THE SYSTEM. IF A RECEIVER-DEHYDRATOR IS LEFT OPEN TO THE ATMOSPHERE IT SHOULD BE REPLACED DUE TO THE LOSS OF EFFECTIVENESS OF THE DRYING COMPOUND IT CONTAINS.

CAUTION: A NEW RECEIVER-DEHYDRATOR SHOULD BE OPENED AND CONNECTED TO THE SYSTEM ONLY WHEN READY TO CHARGE THE SYSTEM WITH REFRIGERANT.

CAUTION: RECOMMENDED TORQUE VALUES MUST BE USED ON ALL FLARE FITTING AND O-RING JOINTS. SEE "CHART 3".

CAUTION: IF AIR CONDITIONING REFRIGERANT LINES OR SYSTEM IS OPENED, LINES AND FITTINGS MUST BE CAPPED AND SEALED IMMEDIATELY TO PREVENT DIRT AND OTHER CONTAMINANTS FROM ENTERING THE SYSTEM. (DO NOT PUT A PLUG INTO THE HOSES OR FITTINGS.)

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS PROHIBIT THE RELEASE OF REFRIGERANT INTO THE ATMOSPHERE. SPECIAL EQUIPMENT IS REQUIRED WHEN TESTING, DISCHARGING, OR CHARGING THE SYSTEM.

CHART 3
RECOMMENDED TORQUE SPECIFICATIONS

ALUMINUM TUBING		
Metal Tube O.D.	Thread and Fitting Size	Ft./Lb.
1/4	7/16	5-7
3/8	5/8	11-13
1/2	3/4	15-20
5/8	7/8	21-27
3/4	1-1/16	28-33

FLARE CONNECTIONS			O-RING CONNECTIONS		
Tube OD	Thread size	Ft./Lb.	Tube OD	Thread size	Ft./Lb.
3/8	5/8	18-20	3/8	5/8	11-13
1/2	3/4	36-39	1/2	3/4	15-20
5/8	7/8	52-57	5/8	7/8	21-27

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS REQUIRE THAT AIR CONDITIONING SYSTEM REPAIRS BE ACCOMPLISHED BY A QUALIFIED SHOP WITH APPROPRIATELY TRAINED PERSONNEL.

E. Servicing the System (with a Charging Stand)

CAUTION: MINERAL OIL AND PAG ARE NOT COMPATIBLE. USE A SEPARATE MANIFOLD TEST SET AND / OR TEST/CHARGING STAND AND RECOVERY SYSTEM FOR EACH REFRIGERANT TYPE.

CAUTION: USE RECOVERY UNIT SPECIFICALLY DESIGNED FOR THE TYPE OF REFRIGERANT USED IN THE AIRCRAFT SYSTEM. UNINTENDED AFFECTS MAY OCCUR IF REFRIGERANTS ARE COMBINED.

(1) Discharging (Bleeding/Purging) the System

CAUTION: APPLIES TO ROBINAIR 34700 OR SIMILAR CHARGING/RECOVERY STATION. IF USING DIFFERENT EQUIPMENT, SEE OPERATOR'S MANUAL OF STATION BEING USED FOR DETAILED INSTRUCTIONS FOR DISCHARGING SYSTEM.

This procedure is for a Robinair 34700 or similar charging stand/recovery station, see "Figure 3". Required only if the system contains refrigerant.

NOTE: The term "Discharge," as used throughout this section, in no sense implies or suggests discharging refrigerant to atmosphere. In all cases when discharging, an environmentally approved refrigerant recovery station is to be used.

(a) Gain access to service valves by removing rear access panel.

(b) Remove protective caps from access valves.

(c) Connect high side (red) hose to air conditioner high side service valve. On systems equipped with quick disconnect connections, open coupler valve.

(d) Connect low side (blue) hose to air conditioner low side service valve. On systems equipped with quick disconnect connections, open coupler valve.

(e) Check the low side gauge (GAUGE 1) and high side gauge (GAUGE 2) to determine that there is pressure in the system. If there is no pressure, there is no refrigerant in the system to recover.

(f) Check that the oil drain valve is closed.

(g) Open both the low side and high side valves on control panel.

(h) Open the red GAS (vapor) valve and the blue LIQUID valve on the charging station's refrigerant tank.

(i) Slowly open the oil drain valve to see if system oil separator contains oil. If it does, let oil drain into the oil drain bottle (located at the bottom of the rear side of the charging station) until separator is empty.

(j) Close the oil drain valve. Dispose of collected oil in an environmentally accepted manner. Return collection bottle to its place on the charging stand.

(k) Plug unit into a proper voltage outlet. Turn MAIN POWER switch ON.

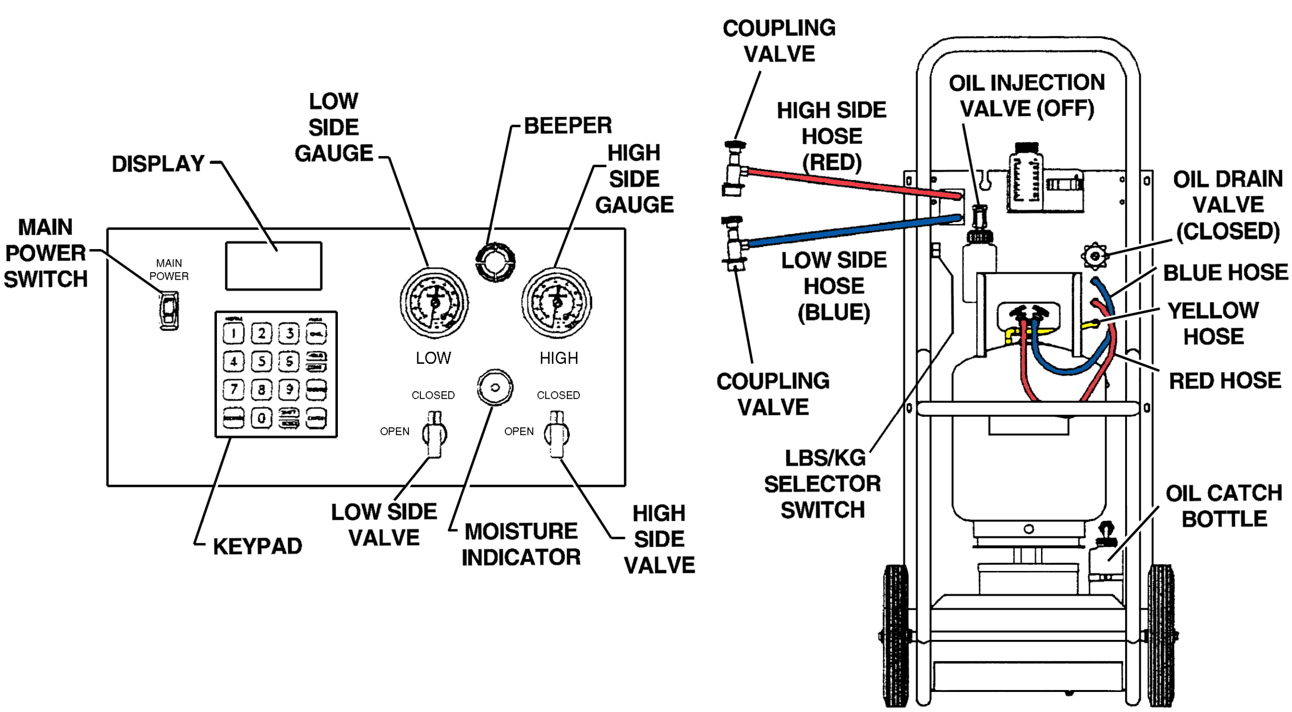
(l) Press the RECOVER key on charging station keypad.

(m) To assure complete recovery of refrigerant:

1) Wait 5 minutes. Observe pressure gauges for a rise above zero.
2) If a rise occurs, press the HOLD/CONT key.
3) Repeat as necessary until system maintains pressure for two minutes.

(n) Slowly open oil drain valve. Drain oil into the oil catch bottle. When all recovered oil has been completely drained, close oil drain valve.

NOTE: Drain oil separator after each job. Display will indicate OIL (OUNCES) or OIL (GRAMS) as a reminder.



Robinair 34700 Charging Stand
Figure 3

(o) Measure the amount of oil in the catch bottle. The same amount of new oil must be added to the system before charging the system.

(p) To enter diagnostic mode, simultaneously press the SHIFT/RESET and ENTER keys. To display amount of refrigerant recovered by the unit, press the 3 key. The panel display will read the amount of recovered refrigerant in pounds or kilograms.

(q) Simultaneously press the SHIFT/RESET and ENTER keys to clear internal counter. Press SHIFT/RESET to return to the main menu.

(2) Evacuating the System

If the system has been operated in a discharged condition or anytime the system has been open to atmospheric pressure, the receiver-dehydrator must be replaced and the system evacuated to remove any trapped air and moisture which has entered it. A vacuum pump capable of pulling 29 inches of mercury or better should be used. As the pressure in the air conditioning system is lowered, the boiling temperature of the water (moisture) that may be present is also lowered. This then forces any moisture, in the form of water vapor, out of the system. Chart 4 demonstrates the effectiveness of moisture removal under a given vacuum.

(a) Using a Kent Moore J23500 or similar charging stand: (See "Figure 4".)

1) Turn aircraft power OFF before beginning.

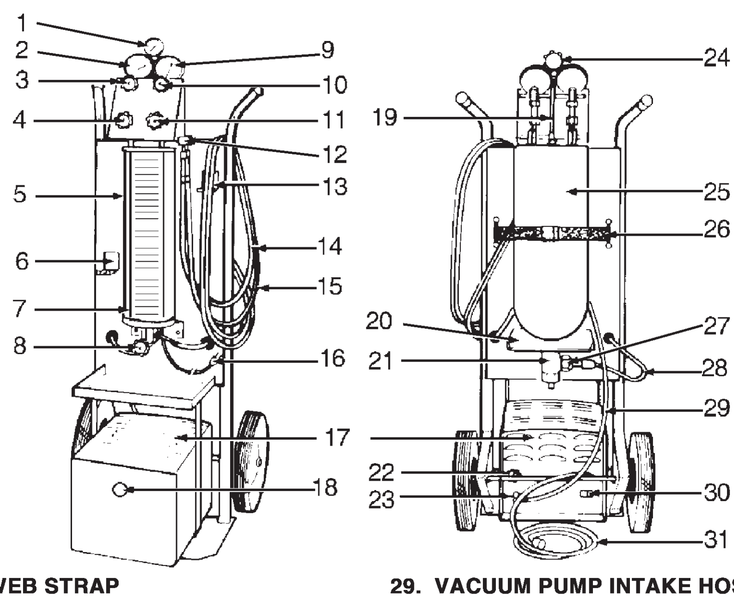
2) Remove the access panel at the rear of the cabin floor to gain access to the service valves on the manifold assembly.

3) Remove the protective caps from the high and low side service ports on the manifold assembly.

4) Close all valves on the charging stand.

5) Remove the protective cap from the vacuum pump outlet.

1. CYLINDER PRESSURE GAUGE
2. COMPOUND GAUGE
3. VALVE, LOW PRESSURE CONTROL
4. VALVE, VACUUM CONTROL
5. CHARGING CYLINDER
6. BRACKET
7. SIGHT GLASS
8. CYLINDER BASE VALVE
9. HIGH PRESSURE GAUGE
10. VALVE, HIGH PRESSURE CONTROL
11. VALVE, REFRIG. CONTROL
12. CHARGING LINE HOSE HOLDER
13. BRACKET
14. LOW PRESSURE CHARGING LINE
15. HIGH PRESSURE CHARGING LINE
16. HEATING ELEMENT PLUG
17. VACUUM PUMP
18. OIL FILL LOCATION
19. NECK ASSEMBLY
20. REFRIGERANT DRUM SUPPORT
21. REFRIGERANT DRUM VALVE
22. VACUUM PUMP VALVE
23. VACUUM PUMP EXHAUST PORT
24. TOP CYLINDER VALVE
25. REFRIGERANT DRUM



26. WEB STRAP
27. REFRIGERANT DRUM REDUCER
28. CHARGING CYLINDER HOSE
29. VACUUM PUMP INTAKE HOSE
30. VACUUM PUMP SWITCH
31. VACUUM PUMP POWER CORD

Kent Moore J23500 Charging Stand
Figure 4

6) Connect the blue and red hoses to the service ports (See “Figure 1” on page 21506 and “Figure 2” on page 21508).
7) Start the vacuum pump.
8) Open the valve on the vacuum pump. Open the low pressure control valve and the vacuum control valve on the charging stand.
9) After five minutes of pump operation, the high pressure gauge should indicate slightly below zero.
   a) If it doesn’t, stop the pump and eliminate the blockage in the system replacing the faulty component, then repeat steps (1) - (8).
   b) If it does, open the high pressure control valve on the charging stand and continue to evacuate the system.
10) Operate the vacuum pump for fifteen minutes, or until the compound gauge indicates 24 to 26 in. Hg. whichever occurs first.
11) Close the low pressure control valve and the high pressure control valve on the charging stand. Stop the vacuum pump and observe the compound gauge. If the gauge rises at a rate faster than 1 in. Hg. in 5 minutes, there is a leak in the system. Locate and fix the leak. Repeat the evacuation steps above.
12) Open the low pressure control valve and the high pressure control valve on the charging stand. Continue pumping and hold the system pressure below 26 in. Hg. for a minimum of 30 minutes. All the pumping time specified above may be included in the 30 minutes provided that no leaks or blockages are noted, and provided that the system is not opened by removal or disconnection of components.
13) Close the low pressure control valve, the high pressure control valve and the vacuum control valve. Stop the vacuum pump and perform the charging procedure immediately.

CHART 4
USING VACUUM TO EVACUATE MOISTURE

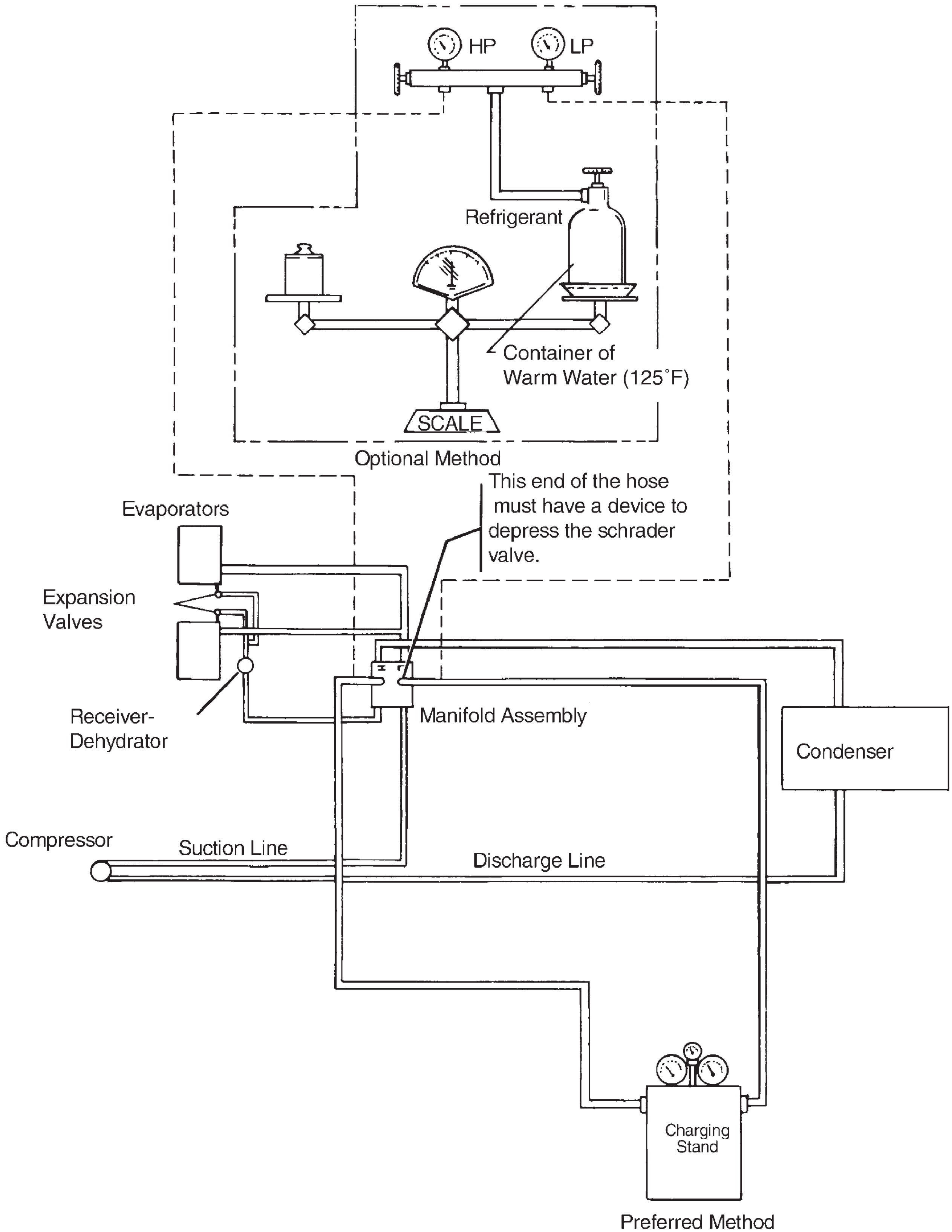
	System Vacuum	Boiling Point of Water (°F)
COMPOUND GAUGE READING IN INCHES OF MERCURY VACUUM	27.95	101
	28.74	84
	29.53	52
	29.76	29
	29.84	15
	29.88	1

NOTE: Compound gauge reading will be approximately one inch lower, numerically, for each 1000 feet elevation above sea level.

(b) Using a Robinair 34700 or similar charging/recovery stand: (See “Figure 3” on page 215014 and “Figure 6” on page 215021.)

Turn aircraft power OFF before beginning.

1) Remove access panel at rear of cabin to gain access to service valves.
2) Remove protective caps from the high and low side service ports on the evaporator unit.
3) Close both the low side and high side valves on charging unit.
4) Connect the blue and red hoses to the service ports (See “Figure 1” on page 21506 and “Figure 2” on page 21508). On systems equipped with quick disconnect connections, open coupler valves.
5) Open blue (low side) valve (1) on unit’s control panel.
6) Open both the red GAS (vapor) valve and the blue LIQUID valve on the tank.
7) Program the length of evacuation time.
    a) Press the VACUUM key on control panel key pad.
    b) Display will show unit is in VACUUM mode.
    c) Refer to operator’s manual for further detail.
8) Enter the required time in minutes and seconds (30:00 minutes minimum) by pressing appropriate keys and then ENTER on keypad. The display will show selected time in minutes and seconds. Example: one hour and fifteen minutes (1:15) would be entered as 7500. The display will show 75:00. Thirty minutes is entered as 3000. the display will show 30:00.
9) To start the vacuum pump press the VACUUM key on keypad again.
10) Vacuum sequence will continue for the programmed time. Digital display will then show CPL, indicating that the evacuation is completed.
11) If, after 5 minutes of pump operation, the RED gauge does not indicate a little below zero:
    a) Stop the pump by pressing the 1 key or the SHFT/RESET key.
    b) Eliminate blockage in the system by replacing faulty parts.
    c) Repeat steps (1) through (8) above.
12) After 5 minutes of pump operation, when RED gauge indicates a little below zero, open red (high side) valve (2), and continue evacuation.
    a) System vacuum should attain 24 to 26 inches of mercury (in. Hg.) in 10 to 15 minutes.
    b) Allow pump to hold system vacuum at or below 26 in. Hg. for a minimum of 15 minutes.
    c) Failure to achieve or hold a vacuum at or below 26 in. Hg. indicates a leak in the system. Perform Leak Detection, below, to locate and repair all leaks.
13) When evacuation sequence has continued for the programmed length of time and the panel display reads CPL (complete), close both the low side valve (1) and the high side (2) valves.
14) Perform charging procedure immediately. (See “Charging the System” on page 215020.)



Charging Hookup
Figure 5

(3) Leak Detection

See “Figure 3” on page 215014 and “Figure 4” on page 215015.

(a) Using Nitrogen

1) Prepare aircraft so as to provide access to all air conditioning components and fitting connections being leak checked.
2) Ensure that aircraft and ground power are OFF or disconnected.
3) Close all valves on the charging station.
4) Remove the protective caps from the high and low side service ports on the manifold assembly.
5) Connect the blue and red hoses to the service ports as shown in “Figure 5”.
6) Slowly pressurize the system to 200 psig max. with nitrogen and turn off the nitrogen source. Ensure gages used have a 0-400 psig range.
7) Monitor pressure on the charging station gauge for 30 minutes. A leak-free system will maintain the 200 psig pressure for 30 minutes.
8) If there is no pressure drop for 30 minutes, slowly release nitrogen pressure and disconnect the nitrogen source from the evaporator assembly. Re-install the pressure switch per drawing requirements. The leak check procedure is completed.
9) If there is a pressure drop, find leak(s) by applying a soap solution to all connections.
10) Rework system as required to stop leaks and repeat Leak Detection.
11) When no leaks are detected, immediately evacuate and charge the system per Evacuating the System, above, and Charging the System, below.

(b) Using HFC-134a

A leak may be located while a charging stand is hooked up to the system as follows:

1) Ensure that there is at least one pound of refrigerant in the charging cylinder.
2) Open the high pressure control valve and the refrigerant control valve on the charging stand. Allow one pound of refrigerant to enter the system.
3) Close the high pressure control valve and the refrigerant control valve.

CAUTION: USE A THICK SOLUTION OF SOAP AND WATER TO CHECK FOR LEAKS INSTEAD OF THE PROPANE LEAK DETECTOR THAT IS PROVIDED WITH SOME BRANDS OF CHARGING STANDS.

4) Locate leak(s) using an electronic leak detector designed to detect HFC134a refrigerant. Or, use soap and water in a thick solution.
5) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.
6) Discharge the system.

NOTE: The term “Discharge,” as used throughout this section, in no sense implies or suggests discharging refrigerant to atmosphere. In all cases when discharging, an environmentally approved refrigerant recovery station is to be used.

7) Evacuate the system.
8) Immediately recharge the system as described in “Charging the System” on page 215020.

(4) Charging the System

NOTE: The following assumes the charging stand is already connected to the airplane as described in Evacuating the System, above.

NOTE: A Post-Charging Operational Check is required immediately upon completion of charging. Accordingly, choose a location where engine run-up is allowed and set up the airplane and the charging stand with the airplane headed into the wind.

(a) Using a Kent Moore J23500 or similar charging stand (See “Figure 4” on page 215015 and “Figure 5” on page 215018).

Ensure aircraft power is OFF before beginning.

1) Open the valve at the base of the charging cylinder and fill the charging cylinder with sufficient refrigerant to charge the system. If refrigerant stops filling the cylinder, open the bleed valve at the top of the charging cylinder to relieve head pressure and allow refrigerant to continue filling the charging cylinder.

2) Close the bleed valve and the valve at the base of the charging cylinder.

3) Turn the charging cylinder sight glass to match the pressure reading on the charging cylinder pressure gauge. Keep the sight glass in this position during the remainder of the charging operation.

4) Connect the heating element plug to a 110 volt power outlet.

5) With the low pressure control valve (3) closed, open the refrigerant control valve (11) and the high pressure control valve (10).

6) Allow correct amount of refrigerant (nominally 2.0 lbs.) to enter the high side of system.

NOTE: The maximum refrigerant capacity is 2.75 pounds. The total refrigerant capacity required is determined separately for each system and is the amount that will result in: bubble-free operation at the sight gauge (R12); or a minimum 15° F temperature drop at the evaporators (HFC-134a); as specified in the Post Charging Operational Check.

7) Close the high pressure control (10) and the refrigerant control valve (11).

8) Perform Post-Charging Operational Check, below.

(b) Using a Robinair 34700 charging station or equivalent (See “Figure 3” on page 215014 and “Figure 6”.)

CAUTION: THE FOLLOWING PROCEDURE APPLIES TO ROBINAIR 34700 OR SIMILAR CHARGING STATION. SEE OPERATOR’S MANUAL OF CHARGING STATION BEING USED, FOR DETAILED INSTRUCTIONS FOR CHARGING SYSTEM.

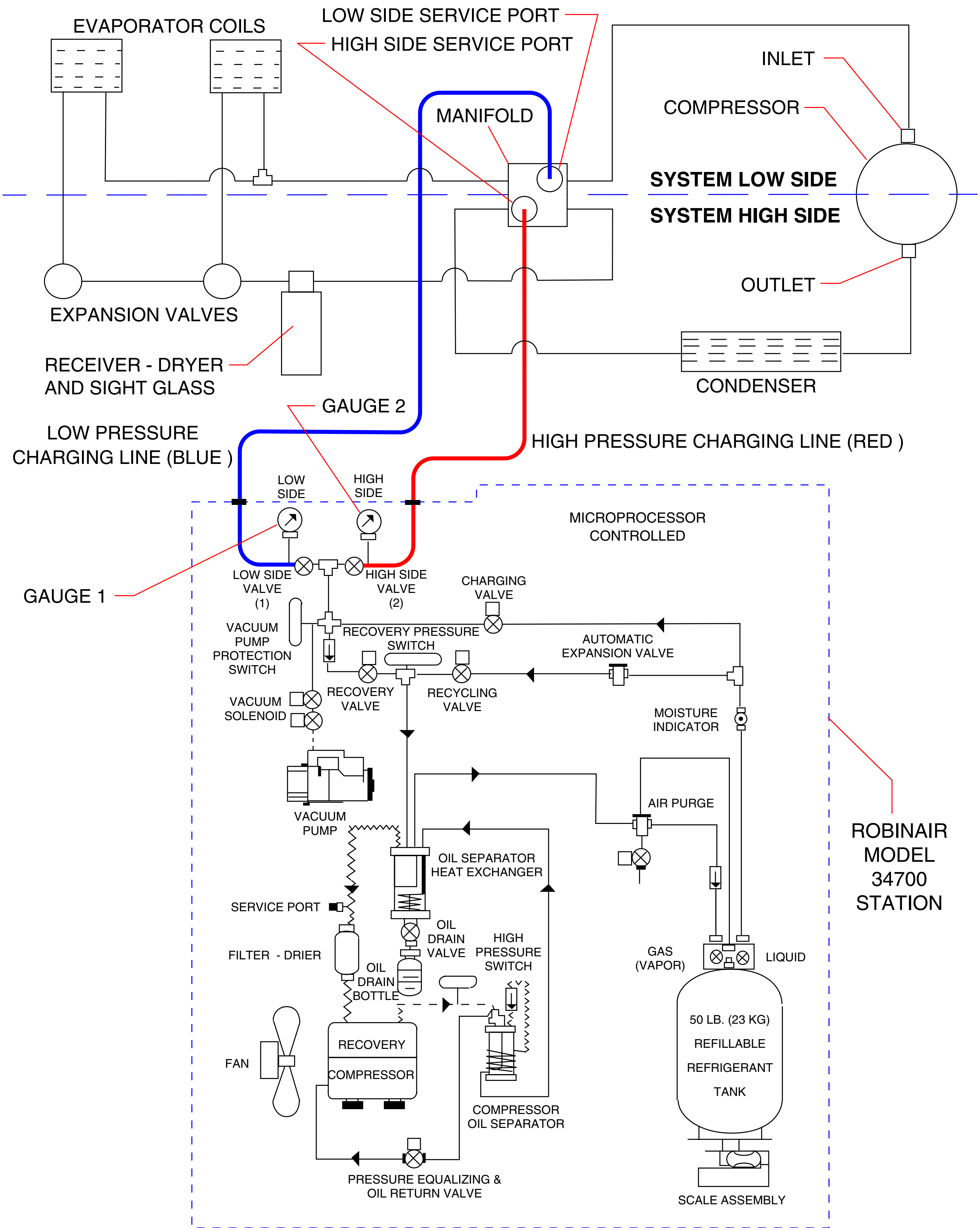
1) Check that main power switch is OFF.

CAUTION: DO NOT PLACE ANY WEIGHT, INCLUDING HANDS AND/OR FEET, ON REFRIGERANT TANK OR SCALE DURING CHARGING PROCESS. ANY WEIGHT DISTURBANCE WILL CAUSE AN INCORRECT TRANSFER OF REFRIGERANT.

2) Check that the LBS/KG selector switch on back of unit is in desired measurement mode.

NOTE: You may enter the amount of refrigerant to be charged when the unit is turned ON. The unit will store the amount in memory until it is turned off.

3) Open the low side (blue) valve on the unit’s control panel.



Robinair 34700 Charging Station Hose Hookup
Figure 6

4) If the messages PROGRAM and CHARGE do not display, press the CHG key to enter the PROGRAM mode.

CAUTION: ADD REFRIGERANT THROUGH THE LOW PRESSURE SIDE ONLY.

NOTE: The maximum refrigerant capacity is 2.75 pounds. The total refrigerant capacity required is determined separately for each system and is the amount that will result in: bubble-free operation at the sight gauge (R12); or a minimum 15° F temperature drop at the evaporators (HFC-134a); as specified in the Post Charging Operational Check.

5) Enter amount of refrigerant (nominally 2.0 lbs.) required to charge the system by pressing the appropriate number keys and ENTER on keypad.
6) To begin charging process, press CHG key on keypad.
   a) The digital display will read AUTOMATIC and show the amount of refrigerant programmed for the charge.
   b) As the solenoid opens, it will make an audible sound.
   c) The display will count down to zero, and display message CPL, when charging is complete.
7) Close low side (blue) valve. Check that the high (red) valve is also closed. Also close coupler valves.
8) Perform Post-Charging Operational Check, below.

(5) Post-Charging Operational Check

NOTE: The following assumes the charging stand is already connected to the airplane as described in Evacuating the System, above.

This procedure can also be used to top off a partially charged system with refrigerant.

(a) Using a Kent Moore J23500 or similar charging stand (See “Figure 4” on page 215015 and “Figure 5” on page 215018 - numbers in parentheses refer to “Figure 4” on page 215015.)

1) Ensure that the low pressure control valve (3) and the high pressure control valve (10) are closed.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE. ENSURE AIRPLANE IS HEADED INTO WIND.

2) Start aircraft engine and bring to normal ramp idle 750 ± 50 rpm. Activate the air conditioning system controls for maximum cooling high blower speed and operate the engine at 1,000 rpm for 15 minutes. Then operate the engine at 2,500 rpm for 15 minutes.
3) Check the sight gauge on the receiver-dehydrator during the engine operation at 1,000 rpm and 2,500 rpm to ensure refrigerant flow.
   a) In R12 systems, bubbles passing the sight gauge indicates additional refrigerant is needed.
   b) Normal HFC-134a operation may appear from clear to foamy. Bubbles passing the sight gauge in a HFC-134a system do not necessarily indicate that additional refrigerant is required. Confirm a minimum 15° F temperature drop between the evaporator inlet air and the evaporator outflow air. If not, then additional refrigerant is required.

   NOTE: Leave cabin door open. If door is closed, cabin ambient temperature will drop and the 15° F temperature delta may be difficult to achieve.

4) If additional refrigerant is required, add it slowly through the refrigerant control valve (11) and the low pressure control valve (3) until the sight glass remains free of bubbles (R12) or a minimum 15° F temperature drop is achieved as described above (HFC-134a). Regulate the flow of refrigerant with the low pressure control valve. Do not allow the compound gauge (2) to exceed a reading of 40 psi.

5) With the engine operating of 1,000-1,500 rpm, the low and high side gauges should indicate as shown in “Chart 5”.

6) With the charge properly established, stop the engine.

7) Close the low pressure control valve (3) and the refrigerant control valve (11).

8) Remove the charging stand. Replace all protective caps, covers, and access panels.

(b) Using a Robinair 34700 charging station or equivalent (See “Figure 3” on page 215014 and “Figure 6” on page 215021 - numbers in parentheses refer to “Figure 6” on page 215021.)

CAUTION: THE FOLLOWING PROCEDURE APPLIES TO ROBINAIR 34700 OR SIMILAR CHARGING STATION. SEE OPERATOR’S MANUAL OF CHARGING STATION BEING USED, FOR DETAILED INSTRUCTIONS FOR CHARGING SYSTEM.

1) Ensure that the low pressure control valve (1) and the high pressure control valve (2) are closed.

2) Ensure coupler valves are open.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE. ENSURE AIRPLANE IS HEADED INTO WIND.

3) Start aircraft engine and bring to normal ramp idle 750 ± 50 rpm. Activate the air conditioning system controls for maximum cooling high blower speed and operate the engine at 1,000 rpm for 15 minutes. Then operate the engine at 2,500 rpm for 15 minutes.

4) Check the sight gauge on the receiver dehydrator during the engine operation at 1,000 rpm and 2,500 rpm to ensure refrigerant flow.

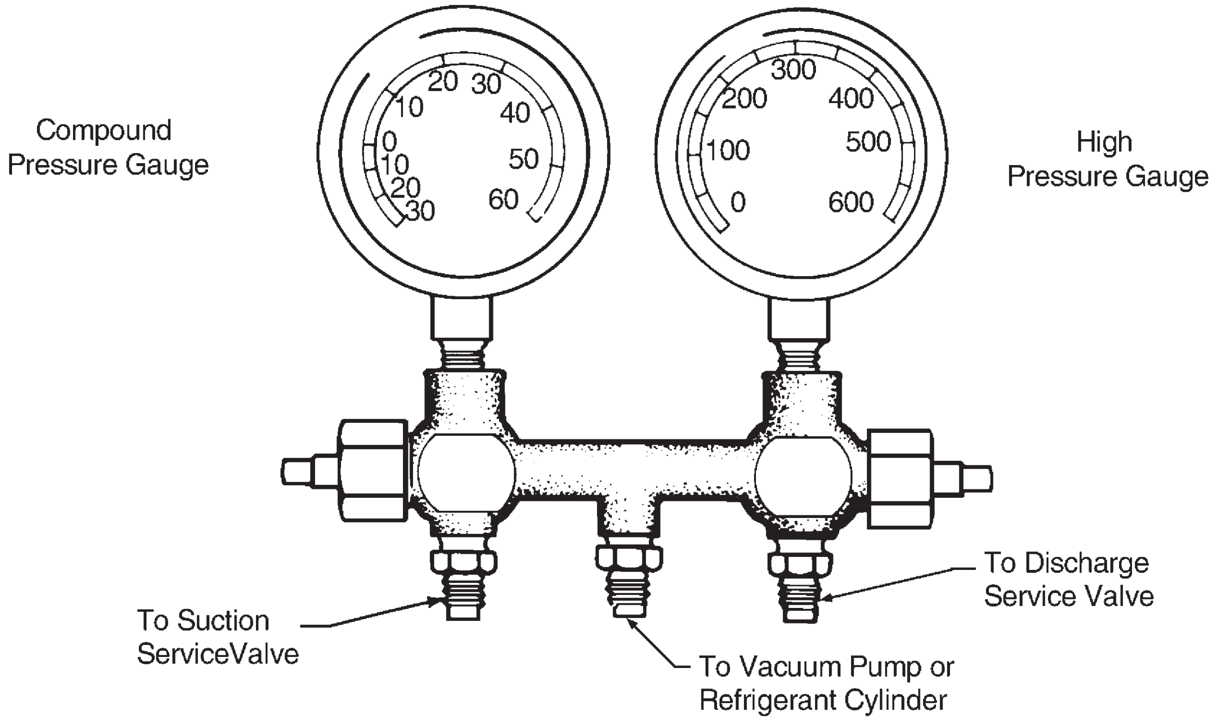
a) In R12 systems, bubbles passing the sight gauge indicates additional refrigerant is needed.

b) Normal HFC-134a operation may appear from clear to foamy. Bubbles passing the sight gauge in a HFC-134a system do not necessarily indicate that additional refrigerant is required. Confirm a minimum 15° F temperature drop between the evaporator inlet air and the evaporator outflow air. If not, then additional refrigerant is required.

NOTE: Leave cabin door open. If door is closed, cabin ambient temperature will drop and the 15° F temperature delta may be difficult to achieve.

CHART 5
AMBIENT TEMPERATURE INDICATION

Gauge	Ambient Temperature	Indication
Low Pressure	All	10 to 35 psig
High Pressure	Up thru 75°F	125 psig min to 175 psig max
High Pressure	Over 75°F	150 psig min to 300 psig max



Test Gauge and Manifold Set
Figure 7

5) If additional refrigerant is required, add it slowly through the low pressure control valve (1) until the sight glass remains free of bubbles (R12) or a minimum 15° F temperature drop is achieved as described above (HFC-134a). Regulate the flow of refrigerant with the low pressure control valve. Do not allow the gauge (1) to exceed a reading of 40 psi.
6) With the engine operating of 1,000-1,500 rpm, the low (1) and high (2) side gauges should indicate as shown in “Chart 5” on page 215023.
7) With the charge properly established, stop the engine and close the coupler valves.
8) Close the low pressure control valve (1).
9) Remove the charging stand. Replace all protective caps, covers, and access panels.

F. Servicing the System (with a Manifold Set)

(1) Test Gauge and Manifold Set

The proper testing and diagnosis of the air conditioning system require that a manifold gauge set (or a charging stand) be attached to the system. This set consists of two gauges mounted to a manifold. One gauge is a high pressure gauge used in the discharge side of the system. The other is a low pressure gauge used in the suction side of the system. The manifold is a device having fittings for both gauges and connection hoses with provisions for controlling the flow of refrigerant through the manifold. (See “Figure 7” and “Figure 8” on page 215026.)

The center port of the manifold set is used for charging or evacuation procedures, or any other service that may be necessary.

Both the high or low side of the manifold have hand shut-off valves. When the hand valve is turned all the way in, in a clockwise direction, the manifold is closed. The pressures on that side of the system will, however, be recorded on the gauge above the hose.

Cracking the hand valve, in the counterclockwise direction, opens the systems to the middle service port of the manifold set. This is desirable only when it is necessary to let refrigerant out of or into the system. (See “Figure 7” and “Figure 8” on page 215026.)

(2) Evacuating the System

See “Figure 9” on page 215027.

(a) Ascertain that all system pressure is released.

(b) Connect the manifold set hoses to the service ports and vacuum pump as shown in “Figure 9” on page 215027.

(c) Close the high side (pressure) and low side (suction) hand valves on the manifold set.

(d) Start the vacuum pump.

(e) Open the low side manifold set hand valve. The low side gauge should show a vacuum.

(f) After five minutes of pump operation the high side gauge should indicate slightly below zero. If it does not, stop the pump and eliminate the blockage in the system by replacing the faulty component, then repeat the previous evacuation steps.

(g) Operate the vacuum pump for fifteen minutes or until the low side gauge indicates 24 to 26 In. Hg. whichever occurs first.

(h) Close the low side hand valve, stop the vacuum pump and observe the low side gauge. If the gauge rises at a rate faster than 1 In. Hg. in 5 minutes, there is a leak in the system. Locate and repair the leak (per the following instructions), then repeat the previous evacuation steps.

(i) With both the low and high side valves open, continue the pumping and hold the system below 26 In. Hg. for a minimum of 30 minutes. All the previous pumping time may be included in the 30 minutes provided that no leaks or blockages are noted, and provided that the system is not opened by removal or disconnection of components.

(j) Close the low and high side hand valves, stop the vacuum pump and perform the charging procedure immediately.

(3) Leak Detection

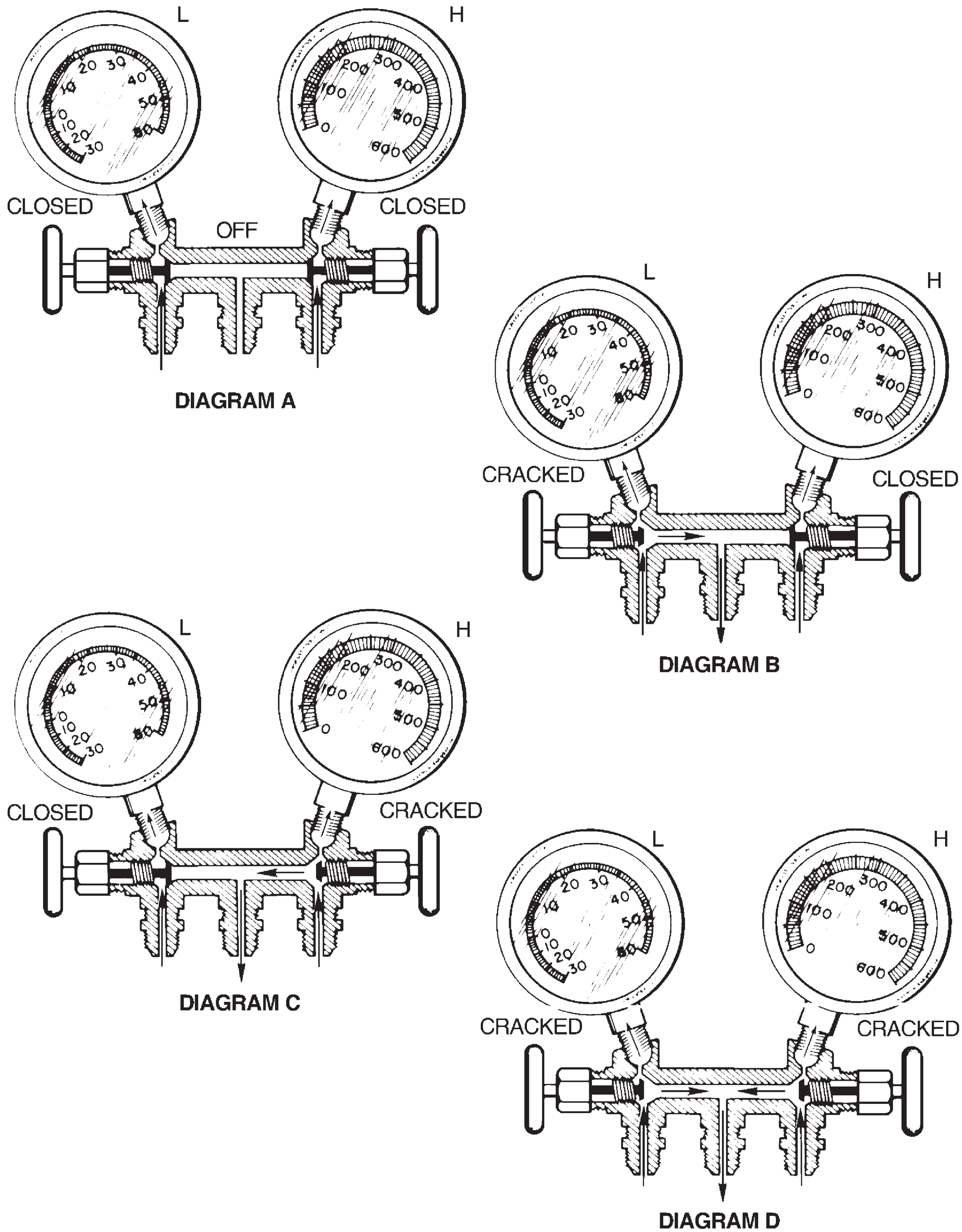
See “Figure 10” on page 215027.

(a) Close both the low side and high side valves on manifold hand set.

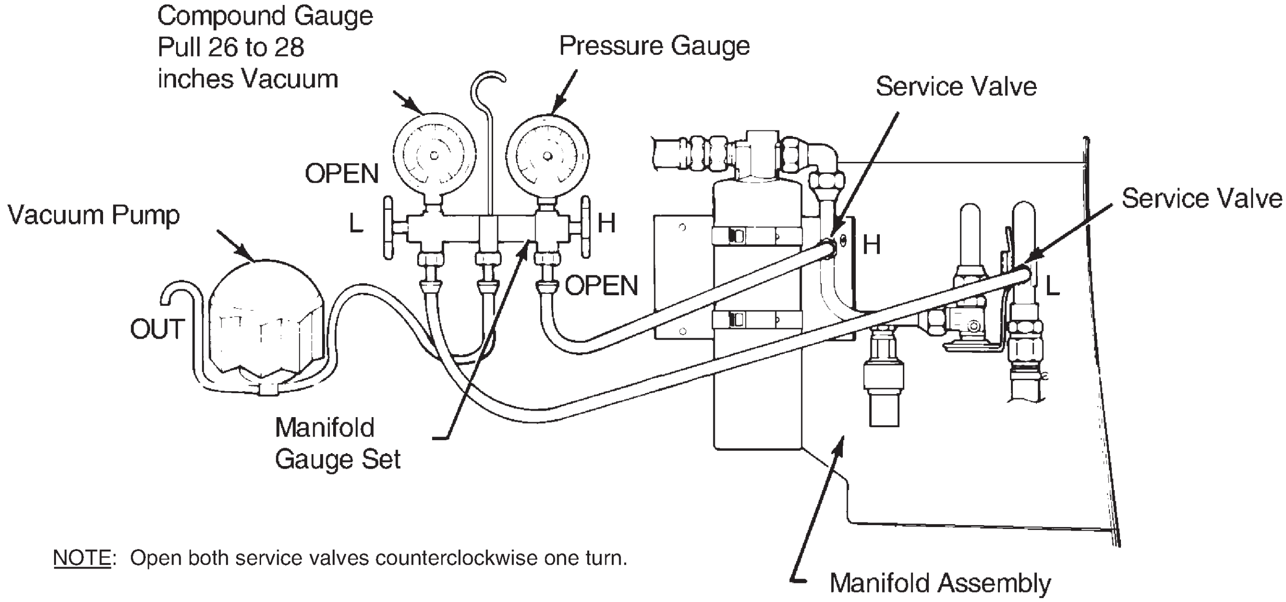
(b) Disconnect manifold hand set middle port (yellow) hose from low side port on charging stand and connect it to a container of R12 or HFC134a refrigerant, as appropriate.

(c) Open refrigerant container service valve.

(d) Open the manifold hand set high side valve until a pressure of 50 psig is reached on low side gauge. Close high side valve.

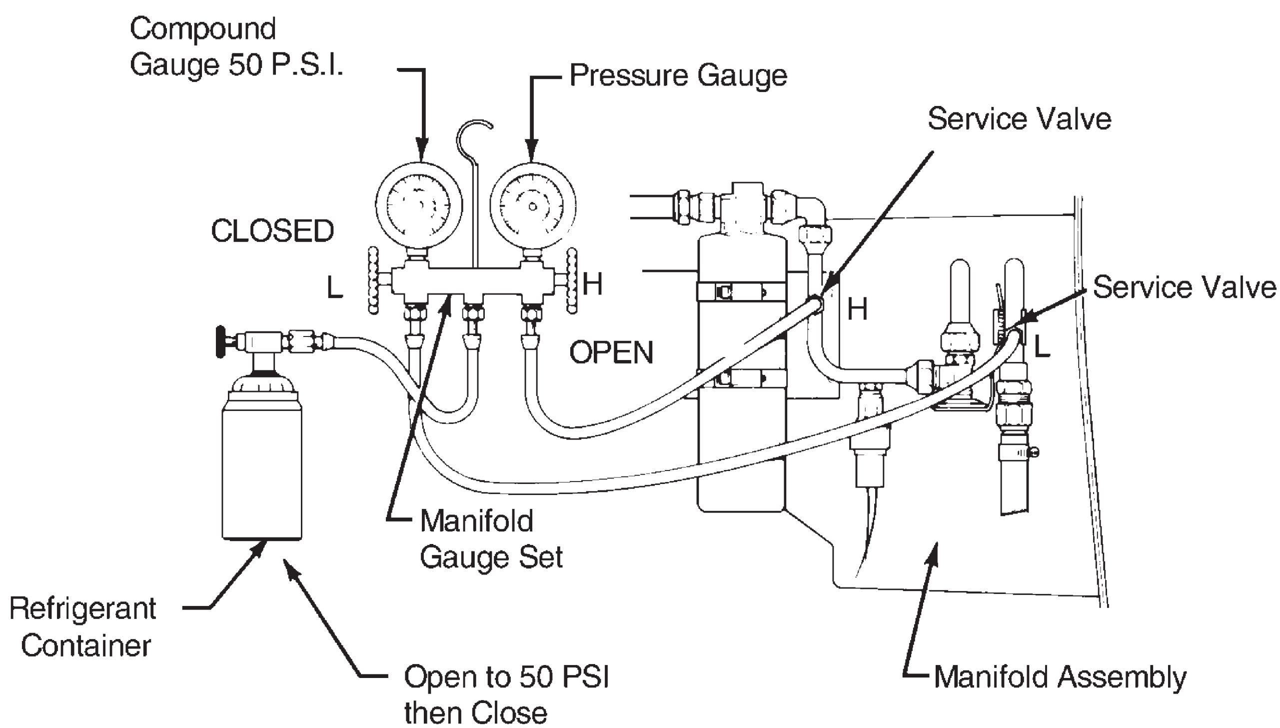


Manifold Set Operation
Figure 8



NOTE: Open both service valves counterclockwise one turn.

Evacuation Hookup
Figure 9



Leak Test Hookup
Figure 10

(e) Locate leak(s) using an electronic leak detector designed to detect R12 and/or R134a refrigerant, as appropriate. Or, use soap and water in a thick solution.

(f) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.

(g) Check that the both high side and low side valves on the manifold hand set are closed.

(h) Close service valve on refrigerant container. Disconnect yellow manifold hand set center hose from refrigerant container.

(i) On systems equipped with quick disconnect connections, close coupler valves. Disconnect manifold hand set red and blue hoses from airplane service ports. Remove manifold hand set.

(j) Recover remaining refrigerant from system using the Robinair 34700 (or other approved) charging station (ref. paragraph 3.C.(1)). Any quantity of oil recovered from aircraft must be measured and an equal amount of new oil (mineral oil or P. A. G., as appropriate) must be added to system before recharging.

(k) On systems equipped with quick disconnect connections, close coupler valves. Disconnect charging/test station from service ports.

(l) Perform evacuation procedure per Evacuating the System, above.

(m) Immediately charge system per Charging the System, above or below.

(4) Charging the System

This method is the least desirable due to the requirement of operating the airplane's engine to run the compressor.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

(a) Keep the system under the vacuum established during the evacuating procedure with both hand valves in the closed position. Disconnect evacuation set-up, if not already disconnected.

(b) Attach a container of the appropriate refrigerant to the manifold set and open the container service valve.

(c) Loosen the center hose at the manifold set until a hiss can be heard. Allow the gas to escape for 2 to 3 seconds, then tighten the connection.

(d) Open the high side manifold set hand valve, observe the low side gauge, then close the high side hand valve. The low side gauge should immediately change from an indication of a vacuum to an indication of pressure. If it does not, the system is blocked, and the blockage must be corrected before proceeding.

(e) Start the engine and operate it at 1000 rpm.

(f) Adjust the airplane air conditioning controls for maximum cooling, high blower speed.

(g) Keep the refrigerant cylinder in an upright position. A slug of liquid refrigerant entering the system would damage the compressor.

(h) Open the low side manifold set hand valve and allow 2.0 pounds of refrigerant in the gas state to enter the system.

(i) Close the low side manifold set hand valve.

(j) Proceed with the Post-Charging Operation Check, below.

(5) Post-Charging Operational Check

This procedure can also be used to top off the system with refrigerant.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

NOTE: Head the airplane into the wind during these checks.

(a) With the manifold set installed, and both hand valves closed;

(b) Start aircraft engine and bring to normal ramp idle 750 ± 50 rpm. Activate the air conditioning system controls for maximum cooling high blower speed and operate the engine at 1,000 rpm for 15 minutes. Then operate the engine at 2,500 rpm for 15 minutes.

(c) Check the sight gauge on the receiver-dehydrator during the engine operation at 1,000 rpm and 2,500 rpm to ensure refrigerant flow.

1) In R12 systems, bubbles passing the sight gauge indicates additional refrigerant is needed.

2) Normal HFC-134a operation may appear from clear to foamy. Bubbles passing the sight gauge in a HFC-134a system do not necessarily indicate that additional refrigerant is required. Confirm a minimum 15° F temperature drop between the evaporator inlet air and the evaporator outflow air. If not, then additional refrigerant is required.

NOTE: Leave cabin door open. If door is closed, cabin ambient temperature will drop and the 15° F temperature delta may be difficult to achieve.

(d) Add additional refrigerant slowly through the low side manifold set hand valve until the sight glass remains free of bubbles (R12) or a minimum 15° F temperature drop is achieved as described above (HFC-134a).

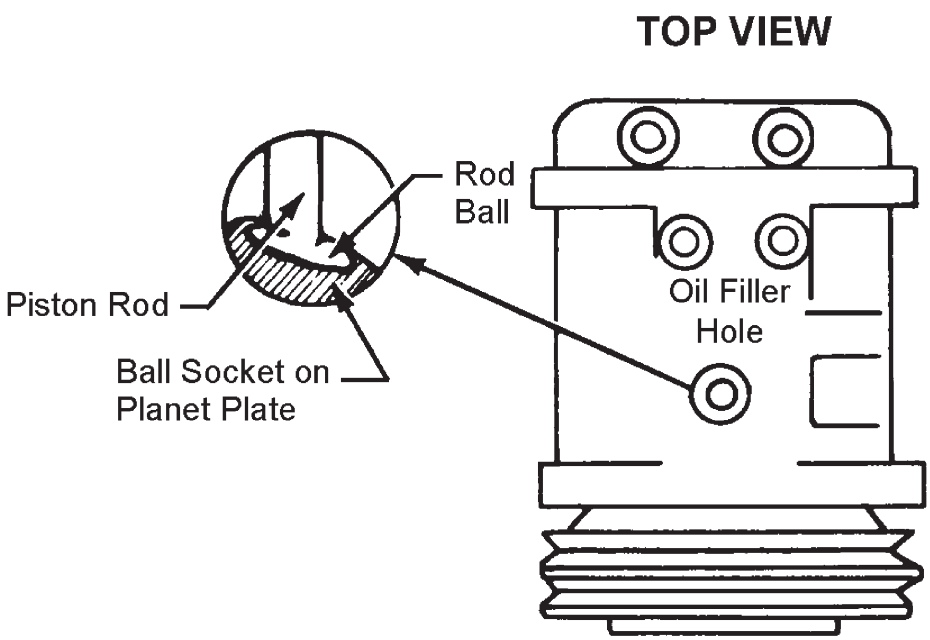
(e) Close the low side hand valve and refrigerant container valve.

(f) With the engine operating at 1,000 to 1,500 rpm, the gauges should indicate as shown in “Chart 6”.

(g) Once the charge is properly established, stop the engine, close the refrigerant container service valve. Remove the manifold set and replace all protective caps, covers, and access panels.

CHART 6
AMBIENT TEMPERATURE INDICATION

Gauge	Ambient Temperature	Indication
Low Pressure	All	10 to 35 psig
High Pressure	Up thru 75°F	125 psig min to 175 psig max
High Pressure	Over 75°F	150 psig min to 300 psig max



Positioning Sanden Compressor Internal Parts
Figure 11

G. Compressor

The engine driven compressor is mounted on the forward, left section of the engine. A V-belt connected to the starter ring gear pulley drives the compressor through a magnetic clutch.

(1) Compressor Service

It is not advisable to service the compressor in the field. It should be done by a qualified shop which has the special equipment and trained personnel required to properly service the unit.

Maintenance to the Sanden compressor is limited to Removal and Installation of Compressor, checking the compressor oil level and replacement of worn drive belt. Contact Sanden International, for special tools and instructions for detailed compressor maintenance. (Refer to the Vendor List in the Introduction Section for the address, and phone number.)

CAUTION: AN IMPORTANT FACTOR IN AIR CONDITIONING SERVICING IS CLEANLINESS. TAKE CARE TO PREVENT DIRT OR FOREIGN MATERIAL FROM ENTERING THE SYSTEM. ALL HOSE AND TUBING ENDS SHOULD BE CAPPED IMMEDIATELY. ANY LUBRICATION REQUIRED IN THE ASSEMBLY OF THE COMPONENTS SHOULD BE REFRIGERANT OIL OF THE TYPE USED IN THE COMPRESSOR.

(2) Removal

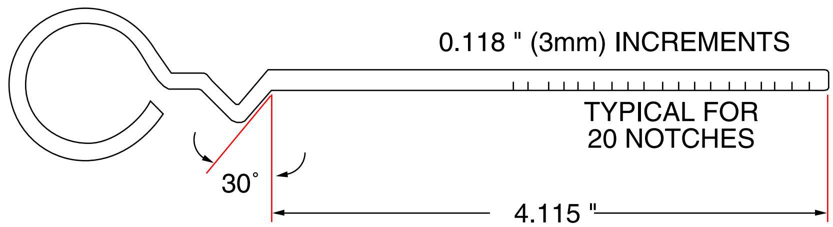
Remove the compressor as follows:

(a) Be sure the air conditioning circuit protector is in the off position.

(b) Remove upper engine cowling and left hand half of nose bowl cowling.

(c) Disconnect the electrical leads to the magnetic clutch on the compressor.

(d) Discharge the air conditioning system to an appropriate environmentally approved refrigerant recovery station. See Servicing the System, “Discharging” on page 215013.



Fabricated Dipstick for Sanden Compressor Oil Check
Figure 12

(e) Remove the suction and discharge lines from the connections on the compressor.

NOTE: All open lines should be capped immediately to prevent dirt and moisture from entering the system.

(f) Remove forward, left hand engine baffles as required.

(g) Loosen compressor attach bolts to relieve belt tension.

(h) Remove belt.

(i) Support compressor and remove the attachment bolts.

(3) Installation

Install the compressor as follows:

(a) Place compressor in mounting brackets and install attach bolts.

(b) Install compressor drive belt. Rotate compressor drive belt. Adjust belt tension (see below).

(c) Check oil level in compressor. (See “Checking Compressor Oil Level”.)

(d) Connect the discharge and suction lines to their respective fittings.

(e) Install engine baffles.

(f) Evacuate and charge the system per instructions given above.

(4) Checking Compressor Oil Level

NOTE: Total system oil capacity is 8.4 oz.

Whenever a system component has been replaced or there is an obvious leak, use the following procedure to check the compressor oil level (after making necessary repairs):

(a) Run compressor for 10 minutes at engine idle rpm.

(b) Recover all refrigerant from the system. See Servicing the System, “Discharging” on page 215013. Be careful not to lose oil.

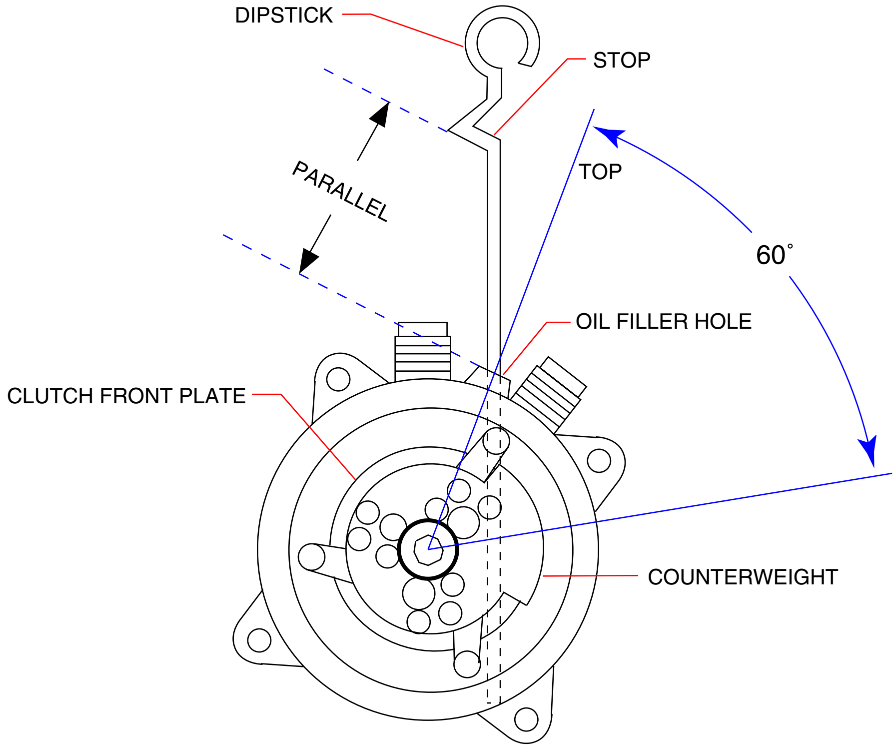
(c) Determine the compressor mounting angle by positioning the angle gauge (Sanden P/N 32448 or a propeller protractor) across the flat surfaces of the two front mounting ears.

(d) Center the bubble and read the mounting angle to the closest degree.

(e) Remove the oil filler plug.

(f) While looking through the oil filler plug hole and observing the internal parts (see “Figure 11”), rotate the counterweight. Stop the counterweight in the position shown in “Figure 13” on page 215032 as the internal parts are moving to the rear of the compressor (discharge stroke).

NOTE: This step clears the internal parts to allow the dipstick to be inserted to full depth.



Sanden Compressor Mounting Angle
Figure 13

(g) Insert the dipstick to its stop position (Refer to “Figure 13”). The stop is the angle near the top of the dipstick.

1) The point of the angle must be to the left if the mounting angle is to the right.

2) The bottom surface of the angle must be flush with the surface of the oil filler hole.

(h) Remove the dipstick and count the increments of oil.

(i) Use “Chart 7” to determine the correct oil level for the mounting angle of the compressor.

(j) If the increments read on the dipstick do not match the table, add or subtract oil to obtain the mid-range value.

(k) Install the oil filler plug, first checking that the sealing O-ring is not twisted. Ensure that the seat and O-ring are clean.

(l) Torque the plug from 6 to 9 foot-pounds (0.8 to 1.2 kg-m). Do not over tighten the plug to stop a leak, If plug leaks, remove it, and install a new O-ring.

(m) Recharge the system per Servicing the System, “Charging the System” on page 215020.

(5) Adjustment of Drive Belt Tension (PIR-89700, Rev. BK.)

NOTE: For adjustment of the alternator belt tension, see 24-30-00.

Adjust the A/C compressor belt tension as follows:

(a) Compressor belt to be installed with-span tension of 120 lbs.

(b) Run engine for a 15 minute period at 1200 RPM

(c) Shut down engine and recheck. Reset tension to 80 lbs. load (if tension falls to 60 lbs.).

CHART 7
SANDEN SD 505 COMPRESSOR MOUNTING ANGLE/OIL LEVEL

Mounting Angle Degree	Acceptable Oil Level In Increments
0	4-6
10	6-8
20	8-10
30	10-11
40	11-12
50	12-13
60	12-13
90	15-16

H. Refrigerant Lines and Routing

NOTE: Before any of the hose couplings are uncoupled, the system must be completely discharged.

(1) Description

NOTE: In S/N's 4636187 and up and S/N's 4692001 and up (i.e. - HFC-134a equipped), hoses are nylon-lined.

The refrigerant lines in this airplane are flexible, high-pressure hoses and should be handled accordingly. The hoses in the power plant area are routed so as to provide maximum protection from heat and abrasion.

From the firewall, the hoses are routed underneath the battery, underneath the left cabin floor back to the manifold assembly, receiver-dehydrator and evaporators. From these components, the lines continue back to the condenser.

(2) Repair

Repairs to hoses and hose fittings at the pressure bulkheads may be accomplished as follows:

(a) S/N's 4636001 thru 4636581 and S/N's 4692001 thru 4692189

Order and install Piper Kit No. 88540.

(b) S/N's 4636582 and up and S/N's 4692190 and up (PIR-PPS20029, Rev. B.)

Aeroquip Nylon Veneer Tube hoses are installed. Leaking hose fittings may be replaced as follows:

1) Required Equipment

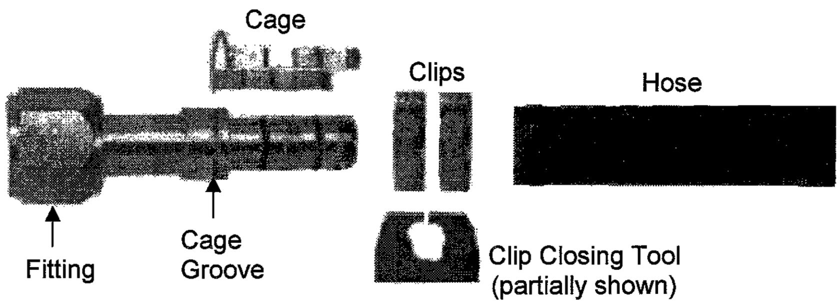
Aeroquip Plier Connecting Tool FT1357. Widely available from various suppliers.

2) Assembly Procedure

a) Consult the appropriate Piper Illustrated Parts Catalog and order the appropriate replacement fitting, cage, and clips. (See "Figure 14" on page 215034.)

b) Prior to installing the fittings on the hose, ensure the hose is cut such that the cut is made square to the hose length.

c) Install two proper sized clips onto the cut end of the hose.



Effectivity
4636582 and up,
4692190 and up,
or with Kit 88540-001 installed

A/C Hose Fitting Installation
Figure 14

d) Lubricate the nipple with a generous amount of Oil, Air Conditioner, HFC-134a, POE (see 91-10-00, Chart 10.).

NOTE: This must be done to lower the force of nipple insertion.

e) Insert the nipple into the hose:

1] There should be no gap between the end of the hose and the shoulder on the nipple.

2] Avoid kinking or damaging the hose while inserting nipple.

f) Snap the cage into the groove on the nipple.

NOTE: When the cage has been correctly installed in the cage groove, the cage will be able to rotate in the groove.

g) Slide the clips over the cage arms and into the channels on each arm.

h) Using Aeroquip tool FT1357, close the clips.

NOTE: The tool should be positioned squarely on the clip connection points and should remain square during the closing of the clip.

(3) Fire Sleeve Installation

See "Figure 15".

On hose assemblies where fire sleeves are required, first install one hose fitting per the "Repair" on page 215033 instructions.

(a) Required Equipment

Aeroquip Plier Connecting Tool FT1357. Widely available from various suppliers.

(b) Assembly Procedure

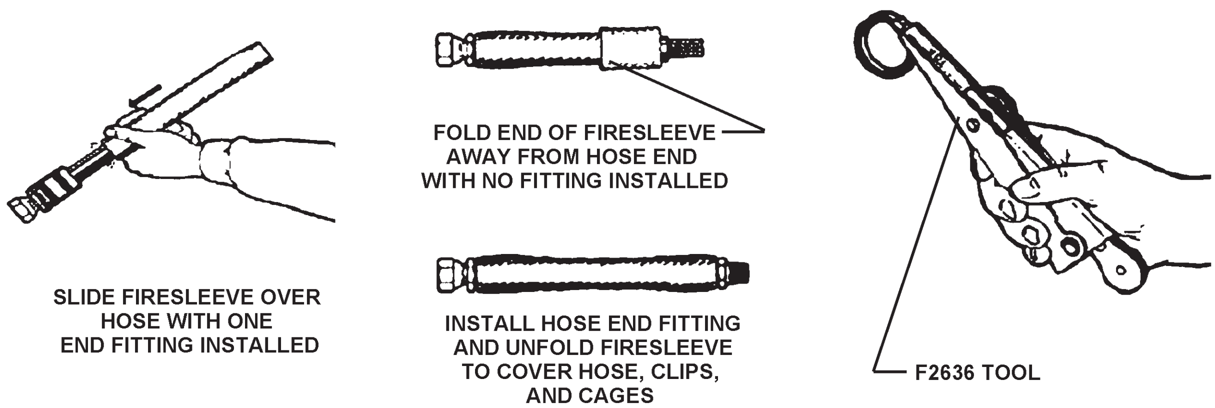
1) Ensure the Fire Sleeve is cut such that the cut is made square to the sleeve length.

2) Slide sleeves over the hose.

3) Peel back the sleeve on the free end of the hose to allow for the assembly of the second fitting.

4) Install the second hose fitting per "Repair" on page 215033 instructions, fold the sleeve back over and center the sleeve on the hose so that it appropriately covers the clips and cages.

5) Insert tail of band clip into Aeroquip tool F2636. Position camp appropriately on the sleeve and draw tight with the tool.



Fire Sleeve Installation
Figure 15

6) Remove tool and cut excess tail end from the clamp.
7) Repeat on other end of hose assembly.

I. Evaporator/Blower Assembly

The two evaporators are located beneath the floor aft of the rib at F.S. 233.87.

(1) Removal
- (a) Discharge the system into an appropriate, environmentally-approved, refrigerant recovery station.
- (b) Remove the refrigerant lines from the inboard side of the evaporator.
- (c) Disconnect the duct from the lower outboard side of the blower.
- (d) Remove the nuts and washers that secure the evaporator/blower assembly to the F.S. 233.87 rib.
- (e) Disconnect the electrical leads from the motor.
- (f) Remove the evaporator/blower assembly.

(2) Installation
- (a) Position the evaporator/blower assembly on the rib at F.S. 233.87 and secure it there with the nuts and washers.
- (b) Connect the electrical leads to the motor.
- (c) Connect the duct to the lower outboard side of the blower.
- (d) Connect the refrigerant lines to the inboard side of the evaporator.
- (e) Charge the system.

J. Expansion Valve

See “Figure 16”.

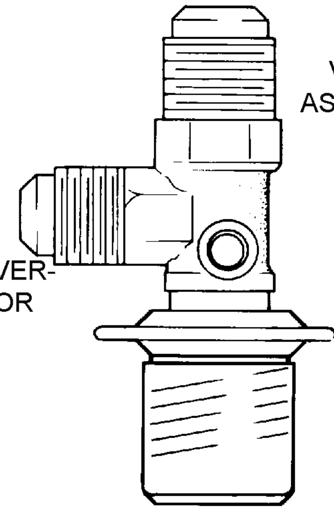
NOTE: If this part is not serviceable, it must be replaced with a new part.

(1) Removal

The expansion valves are located on the upper inboard corners of the evaporator assemblies between the receiver-dehydrator and the evaporator inlet.

(a) Remove the necessary access panels and discharge system into an appropriate, environmentally-approved, refrigerant recovery station.

(b) Uncouple all related tube fittings. (see “Special Servicing Procedures” on page 215011.)



Expansion Valve
Figure 16

(2) Installation

(a) Install the expansion valve in the inlet line of the evaporator core by coupling the related fittings. (In R-12 systems only, seal all couplings with sealant (i.e. - Loctite Refrigerant Sealant) applied to tube flanges only.) Torque fittings per “Chart 3” on page 215013.

(b) Evacuate and charge the system. Check for leaks. (See “Servicing the System (with a Charging Stand)” on page 215013.)

(c) Reinstall any access panels that were previously removed.

K. Condenser

See “Figure 17” on page 215038.

The condenser, blower motor and exhaust duct assemblies are located in the aft fuselage between F.S. 282.00 and F.S. 292.00 (forward of the tailcone access panel).

(1) Removal

(a) Remove the tailcone access panel.

(b) Discharge the air conditioning system.

(c) Disconnect and cap the lines going to the condenser. Cap inlet and outlet ports on the condenser.

(d) Remove the screws which secure the lower end of the duct to the fuselage skin forward of the tailcone access panel.

(e) Remove the bolts, washers, screws and nuts that secure the upper end of the duct to the supports and to the condenser assembly.

NOTE: With the bolts and screws removed from the duct assembly, there is no support for the condenser and it is free to fall.

(f) Carefully position the duct out of the way and remove the condenser.

(2) Installation

(a) Remove caps from lines and ports on condenser.

(b) Position condenser between brackets and align the bolt holes in the supports, condenser and upper end of the duct. Install the bolts, washers, screws and nuts.

(c) Secure the lower end of the duct to the fuselage skin with the screws.

(d) Reconnect the lines to the condenser.

NOTE: Whenever the air conditioning system has been open to the atmosphere, it is recommended that the receiver-dehydrator be replaced.

(e) Evacuate and recharge the system per instructions in this section.

(f) When the system is completely charged, check it for any leaks.

(g) Reinstall the tailcone access panel.

L. Condenser Blower Motor Assembly

See “Figure 17” on page 215038.

The condenser blower motor assembly is located in the condenser exhaust duct. Access to the motor is through the tailcone access panel and the removable access panel on the aft side of the duct assembly.

(1) Removal

(a) Remove the tailcone access panel.

(b) Remove the bolts and washers that secure the duct access panel to the duct, condenser and condenser support. Remove the duct access panel.

(c) Disconnect the electrical leads from the blower motor.

(d) Remove the screws, washers and nuts that secure the motor brackets to the duct.

(e) Lift the blower motor and brackets from the duct.

(f) Remove safety wire, bolts and spacers that secure the motor to the aft bracket.

(g) Remove safety wire, bolts and spacers that secure the motor to the forward bracket.

(2) Installation

(a) Position the blower motor in the forward motor bracket and secure with bolts, spacers and safety wire.

(b) Place aft motor bracket on blower motor and forward motor bracket and secure with bolts, spacers and safety wire.

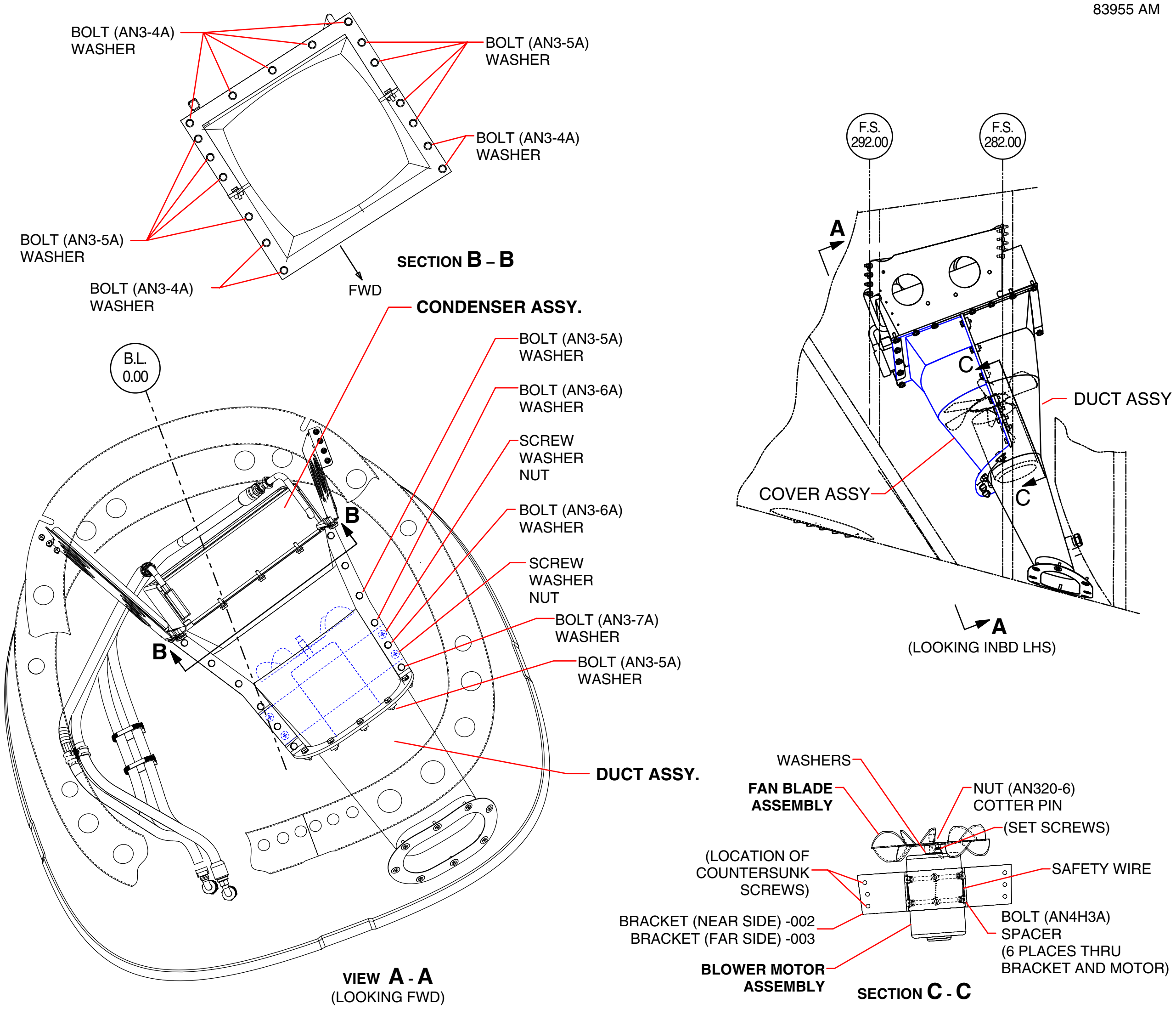
(c) Position blower motor and motor bracket assembly in the duct.

(d) Place screws and washers thru aft and forward motor brackets. Secure to duct with washers and nuts.

(e) Connect the electrical leads to the blower motor.

(f) Secure the duct access panel to the duct, condenser and condenser supports with the bolts and washers.

(g) Reinstall the tailcone access panel.



Condenser Blower Motor Assembly
Figure 17

M. Condenser Blower Motor Fan Blade Assembly

See “Figure 17”.

(1) Removal

(a) Remove Condenser Blower Motor Assembly per Removal, above.

(b) Remove cotter pin and nut securing fan blade to Blower Motor Assembly.

(c) Unscrew two set screws sufficiently to loosen fan blade.

(d) Remove Fan Blade Assembly.

(2) Installation

(a) Apply Loctite 271 to set screws.

(b) Position fan blade on Blower Motor Assembly and washers.

(c) Locate set screws to seat on flat surfaces. Torque set screws to 40–50 in.-lbs.

(d) Using nut and cotter pin, secure fan blade to Blower Motor Assembly.

(e) Reinstall Condenser Blower Motor Assembly per Installation, above.

N. Receiver - Dehydrator

The receiver-dehydrator is located beneath the floor on the left hand side. It is aft of the left evaporator module. The receiver-dehydrator is mounted to the aft side of the frame at F.S. 241.50.

The receiver-dehydrator should be replaced if the air conditioning system has been operating with a leak that would have allowed air to enter the system if the receiver-dehydrator is left open to the atmosphere, it should be replaced due to the loss of effectiveness of the drying compound it contains.

(1) Removal

(a) Discharge the system into an appropriate, environmentally-approved, refrigerant recovery station.

(b) Remove the two refrigerant lines that connect to the top of the receiver-dehydrator. Cap the lines.

(c) In S/N’s 4636187 and up and S/N’s 4692001 and up, disconnect wiring to receiver-dehydrator mounted pressure switches.

(d) Loosen the two clamps that secure the receiver-dehydrator and lift the receiver-dehydrator from its mount.

(2) Installation

(a) Place the receiver-dehydrator in its mount.

(b) Connect the two refrigerant lines to the top of the receiver-dehydrator.

(c) Secure the receiver-dehydrator to its mount by tightening the two clamps around the receiver-hydrator body.

(d) In S/N’s 4636187 and up and S/N’s 4692001 and up, connect wiring to receiver-dehydrator mounted pressure switches.

(e) Service the air conditioning system.

O. Manifold Assembly

The manifold assembly is located behind the receiver-dehydrator and is mounted to the forward face of the frame at F.S. 249.60.

The manifold assembly has the high pressure switch screwed into the high pressure port that goes through the manifold body and the low pressure switch screwed into the low pressure port that passes through the manifold body.

(1) Removal

(a) Discharge the air conditioning system into an appropriate, environmentally-approved, refrigerant recovery station.

(b) Disconnect the four refrigerant lines that go to the manifold assembly.

(c) In S/N’s 4636001 thru 4636186 only, disconnect wiring to manifold-mounted pressure switches.

(d) Remove the two screws that secure the manifold to its mounting bracket and remove the manifold assembly.

(2) Installation

(a) Secure the manifold assembly to its mounting bracket with the two screws. The “H” for stamping should be forward.

(b) In S/N’s 4636001 thru 4636186 only, connect wiring to manifold-mounted pressure switches.

(c) Connect the four refrigerant lines to the manifold assembly.

(d) Service the air conditioning system.

P. Pressure Switches

In S/N’s 4636001 thru 4636186 only, separate high and low pressure switches are located underneath the manifold assembly. See “Condenser Blower Motor Assembly” on page 215037.

In S/N’s 4636187 and up and S/N’s 4692001 and up, the high and low pressure switches have been incorporated into a single housing and are located on the receiver-dehydrator.

In either case, they control the system pressure by energizing or de-energizing the compressor clutch.

(1) Removal

(a) Discharge the air conditioning system into an appropriate, environmentally-approved, refrigerant recovery station.

(b) Gain access to the manifold assembly or receiver-dehydrator.

(c) Remove the electrical connections from the switch.

(d) Turn the switch out of the manifold or receiver-dehydrator body.

(e) Plug the switch hole in the manifold or receiver-dehydrator body.

(2) Installation

(a) Remove the plug from the switch hole in the manifold or receiver-dehydrator body.

(b) Lubricate the switch threads and packing with refrigerant oil and install the switch in the manifold or receiver-dehydrator body.

(c) Service the air conditioning system.

(d) Check operation of the switches.

TEMPERATURE CONTROL

1. Cabin Air Temperature Control Valve

A. Removal

(1) Disconnect the hose from between the heat muff and the duct on the inboard side of the cabin temperature control valve.

(2) Disconnect the duct between the air inlet scoop and the under side of the cabin temperature control valve.

(3) Disconnect the cabin temperature control valve flapper control cable from the cabin temperature control valve.

(4) Remove the four bolts that attach the cabin air temperature valve to the air selector valve (PA-46-350P) or heater duct (PA-46R-350T) (through the firewall).

(5) Remove the valve from the airplane.

B. Installation

(1) Position the cabin temperature control valve in place.

(2) Attach the cabin temperature control valve to the air selector valve (PA-46-350P) or heater duct (PA-46R-350T) through the firewall with four bolts. Ensure that the gasket is installed between the valves and firewall.

(3) Attach the cabin temperature control valve flapper control cable to the cabin temperature control valve.

(4) Connect the duct from the air inlet scoop to the under side of the cabin temperature control valve.

(5) Connect the hose from the heat muff to the duct on the inboard side of the cabin temperature control valve.

C. Rigging

(1) Push cabin temperature control knob in.

(2) Ensure that the air modulation valve is fully open for ram air flow and closed to heat muff hot air flow.

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CHAPTER

22

AUTOFLIGHT

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CHAPTER 22

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
22-List of Effective Pages	1	Mar 25/21
	2	Sep 15/09
22-Table of Contents	1	Nov 30/17
	2	Mar 25/21
22-10-00	1	Nov 30/17
	2	Dec 15/14
	3	Dec 15/14
	4	Dec 15/14
	5	Dec 15/14
	6	Dec 15/14
	7	Dec 15/14
	8	Dec 15/14
	9	Nov 30/17
	10	Nov 30/17
	11	Nov 30/17
	12	Dec 15/14
	13	Nov 30/17
	14	Nov 30/17
	15	Dec 15/14
	16	Nov 30/17
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	26	Dec 15/14
	27	Mar 25/21
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	57	Nov 30/17
	58	Nov 30/17
	59	Nov 30/17
	60	Nov 30/17

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CHAPTER 22 - AUTOFLIGHT

TABLE OF CONTENTS

CHAPTER 22 - AUTOFLIGHT

TABLE OF CONTENTS (continued)

AUTOPILOT

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. General

   A. 1995–1999

      S/N's 4636001 thru 4636247, less 4636160

      Bendix/King Autopilot/Flight Director (AP/FD), manufactured by then Allied Signal (now Honeywell), was installed in these airplanes. Maintenance information for those systems is not included in this manual. Follow the service literature published by the AP/FD equipment manufacturer. This includes mechanical service such as: adjusting bridle cable tension, servo removal and installation, servo clutch adjustments, etc.

      King/Allied Signal/Honeywell technical support, parts support, and service literature can be obtained as shown under Supplementary Publications, Vendor Publications, in the Introduction.

   B. 2000–2010

      S/N's 4636160, 4636248–4636459, 4636461–4636462, 4636481; and 4692001–4692133, 4692141, 4692149, 4692153

      The S-TEC System 55/55X is installed in these airplanes. Maintenance information for this system is provided below.

   C. 2010 and up

      S/N's 4636460, 4636463 and up; 4692134 and up

      The Garmin GFC 700 Automatic Flight Control System (AFCS) is fully integrated within the G1000/G1000 NXi system installed in these airplanes. Maintenance information for this system is provided below.

2. S-TEC System 55 / 55X

   See “Figure 1” on page 22102.

   A. S-TEC System 55 (2000)

      This system was adopted in 2000 and is installed as standard equipment in S/N's 4636160, 4636248 thru 4636313 only.

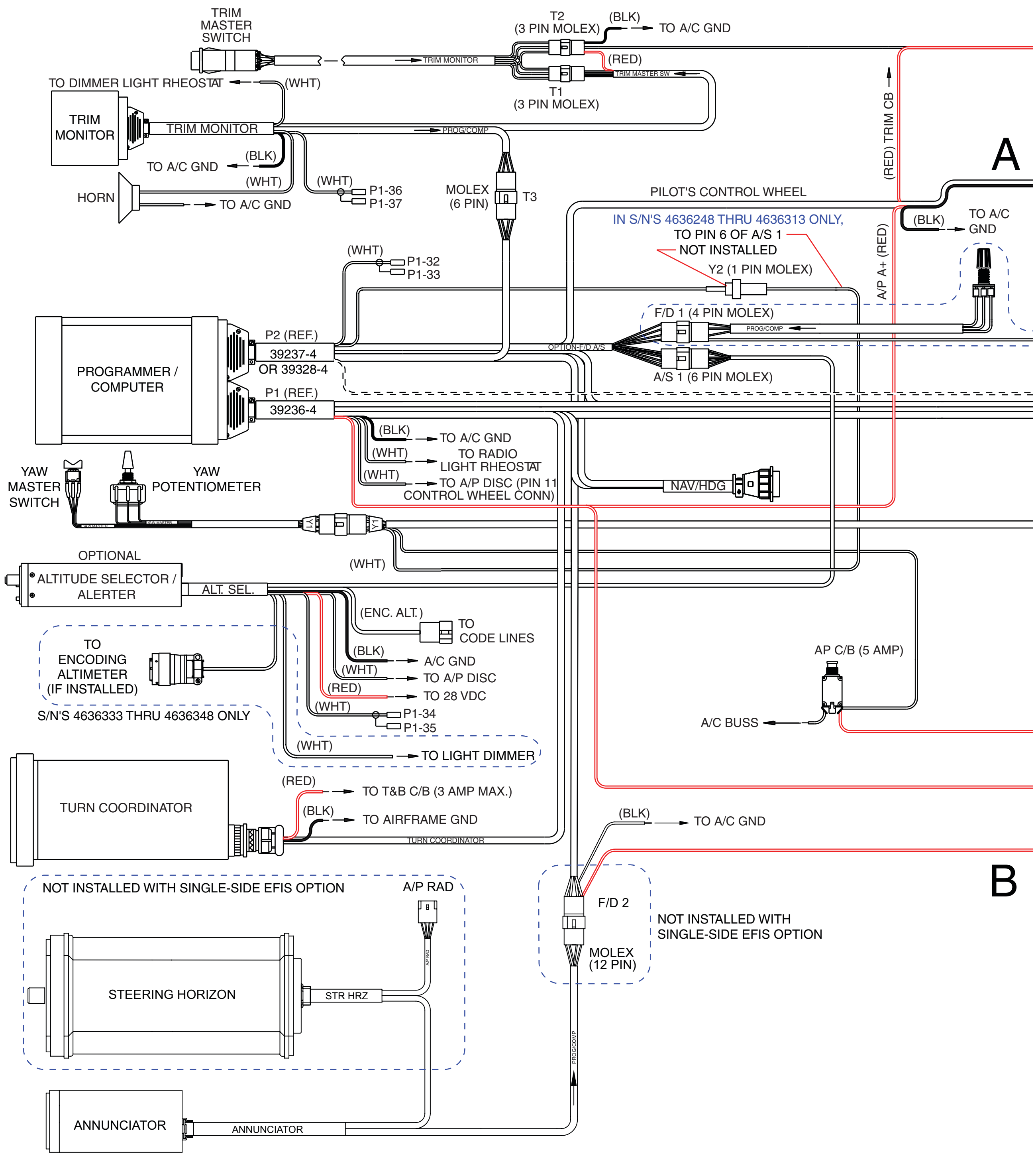
   B. S-TEC System 55X (2002–2010)

      This system was adopted in 2001 and is installed as standard equipment in S/N's 4636314–4636459, 4636461–4636462, 4636481; and 4692001–4692133, 4692141, 4692149, 4692153.

      NOTE: No model year 2001 PA-46-350P airplanes were built.

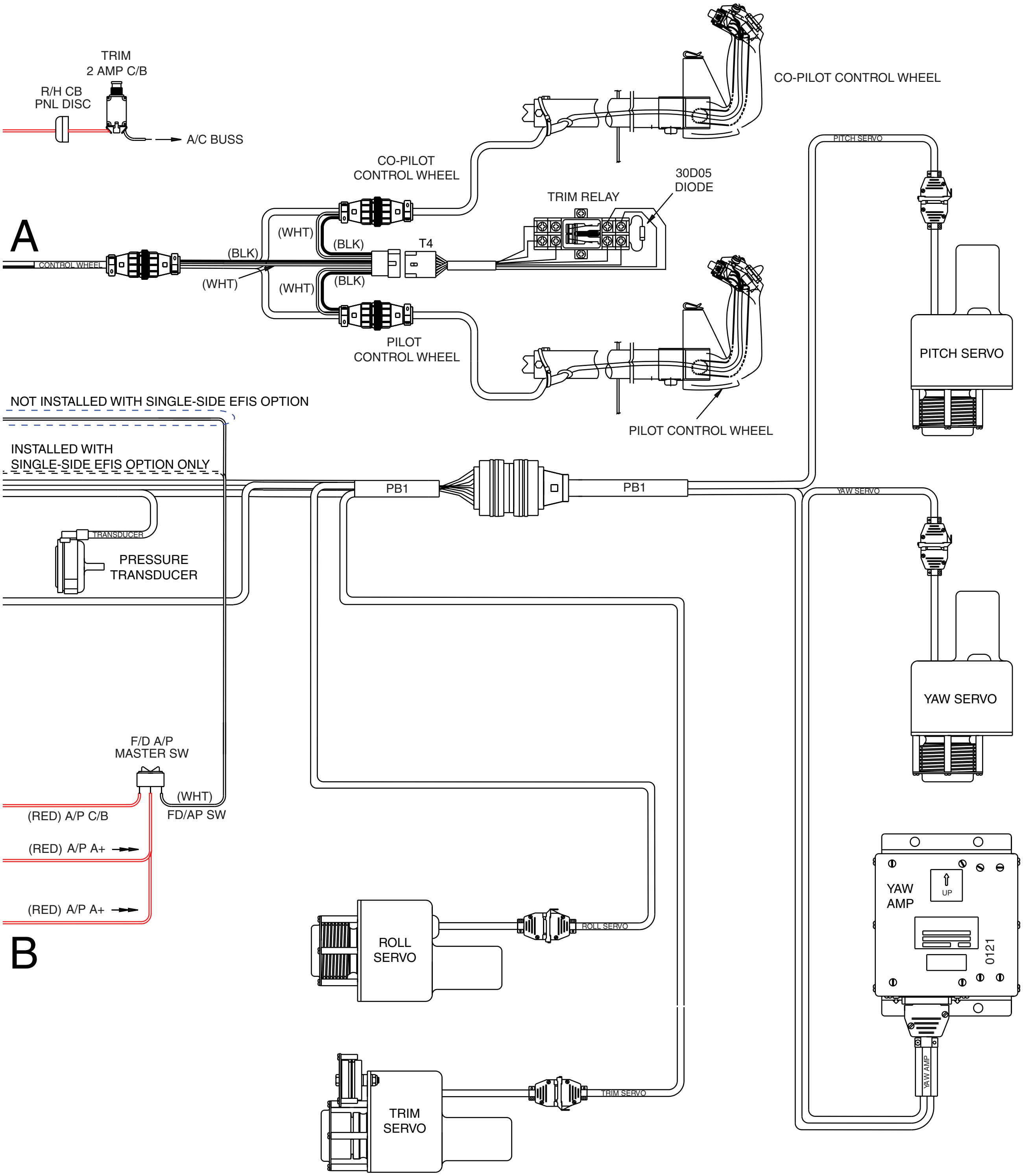
      NOTE: In airplanes equipped with either the optional Meggitt Single Side EFIS (see 34-21-00) or Avidyne Entegra (see 34-22-00); many of the individual components listed below (i.e. - HSI, Deviation Indicators, etc.) are replaced by the Primary Flight Display (PFD). See “Figure 1” on page 22102 and the Avidyne AFM Supplement.

101349 H
101486 E
104585 NEW/E



Effectivity
4636160,
4636248 thru 4636374

S-TEC System 55/55X Installation
Figure 1 (Sheet 1 of 7)



S-TEC System 55/55X Installation
Figure 1 (Sheet 2 of 7)

Effectivity
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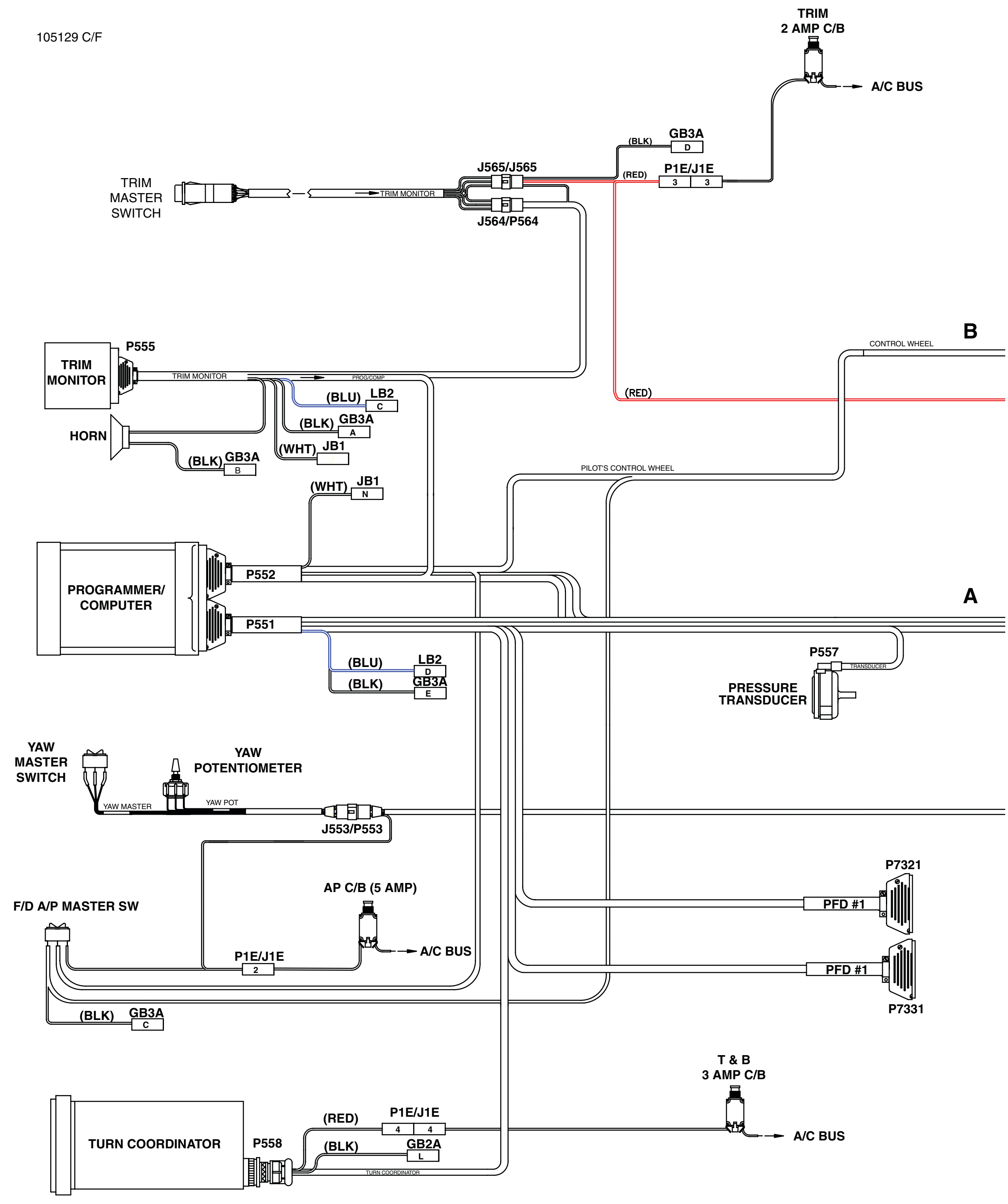
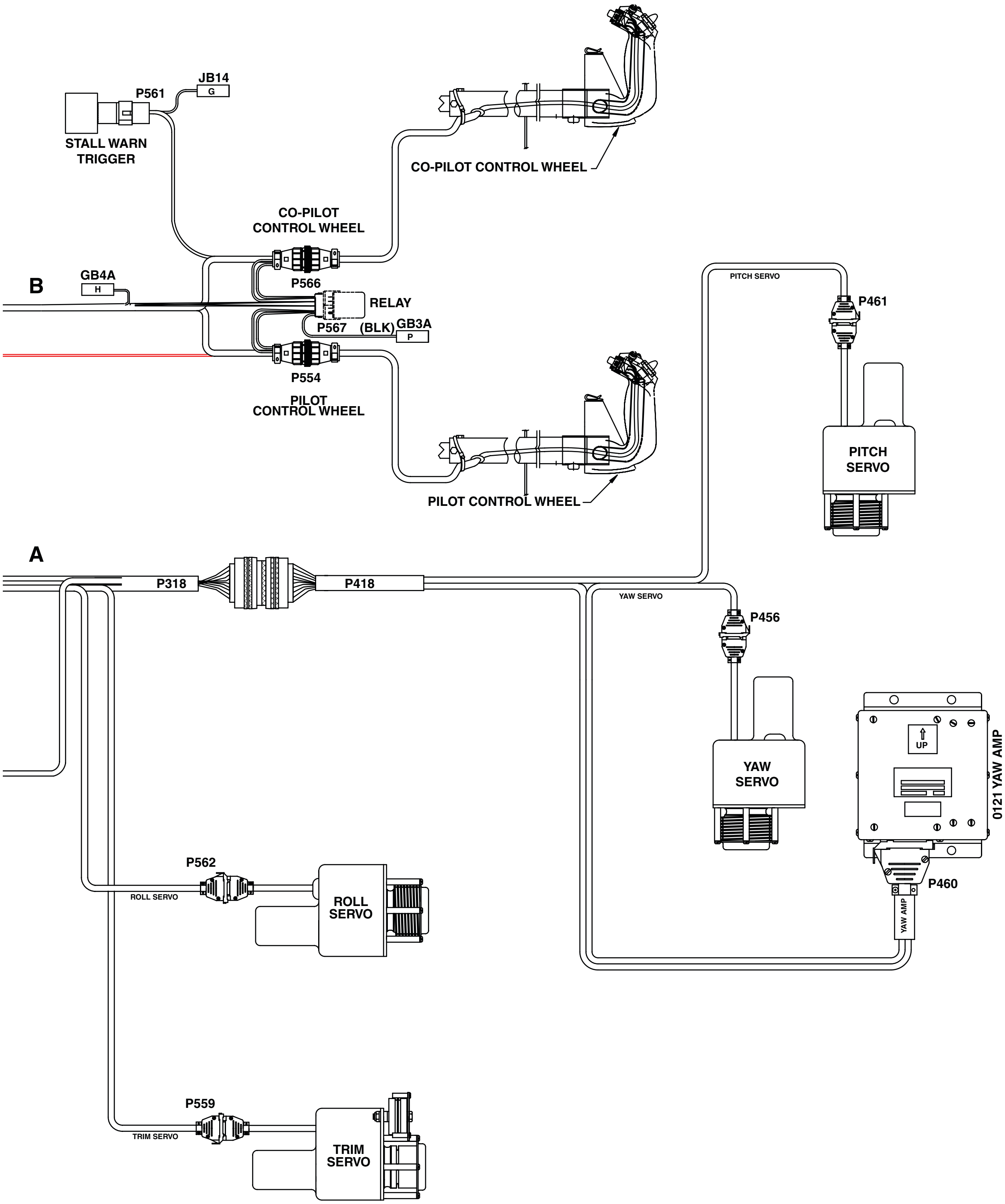
4636248 thru 4636374



S-TEC System 55/55X Installation
Figure 1 (Sheet 3 of 7)

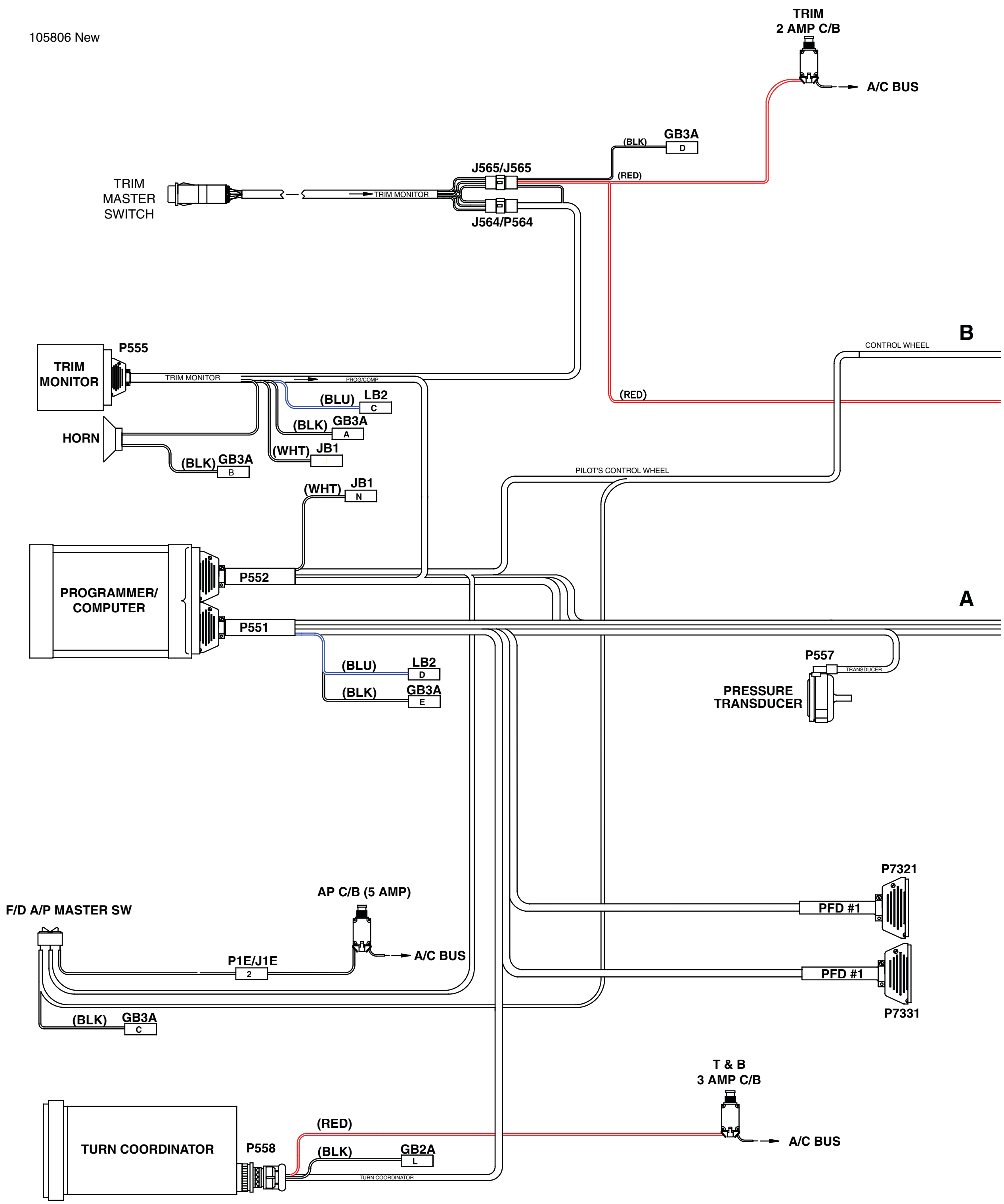
Effectivity
4636375–4636459
4636461–4636462, 4636481



S-TEC System 55/55X Autopilot/Flight Director Installation
Figure 1 (Sheet 4 of 7)

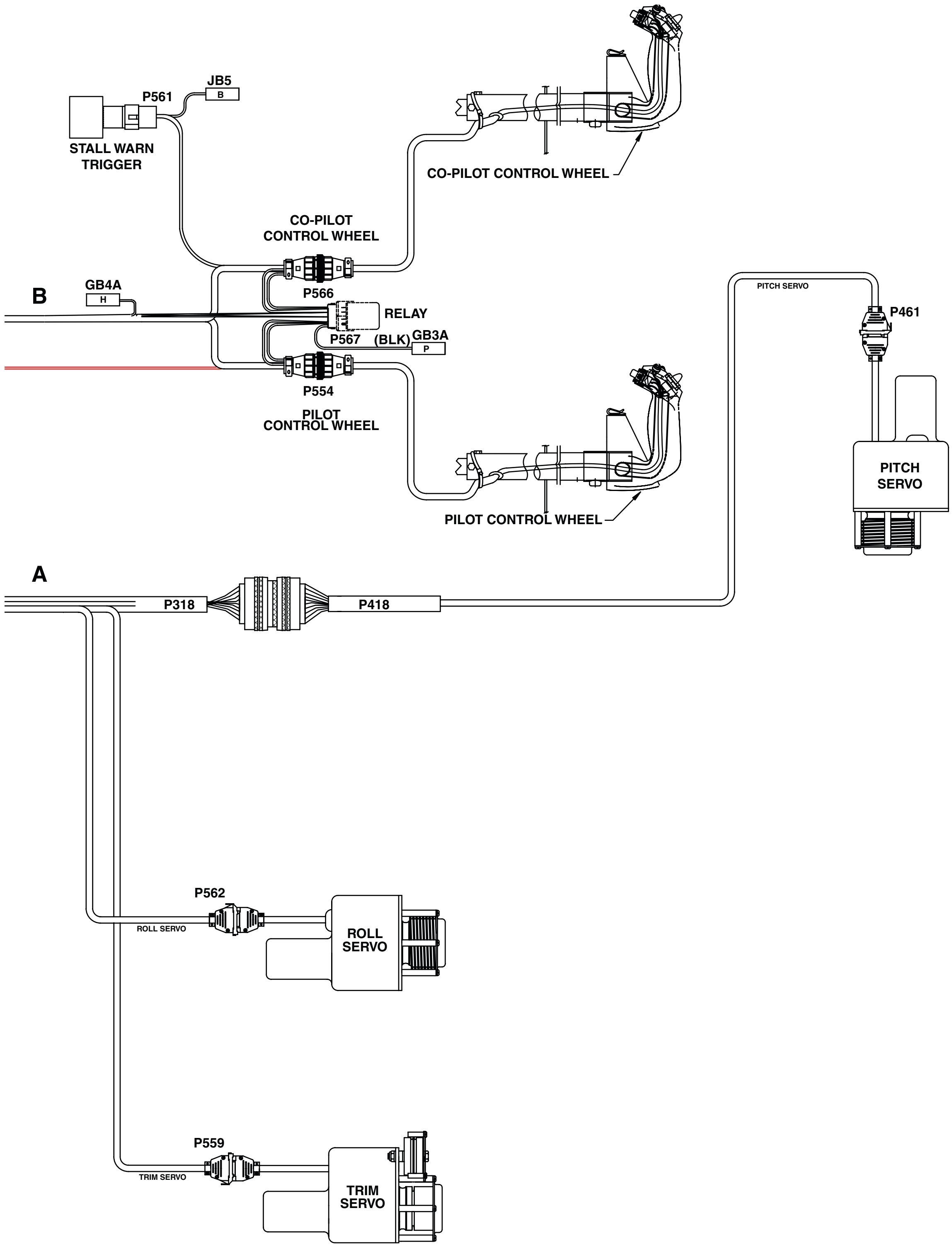
Effectivity
4636375–4636459
4636461–4636462, 4636481

105806 New



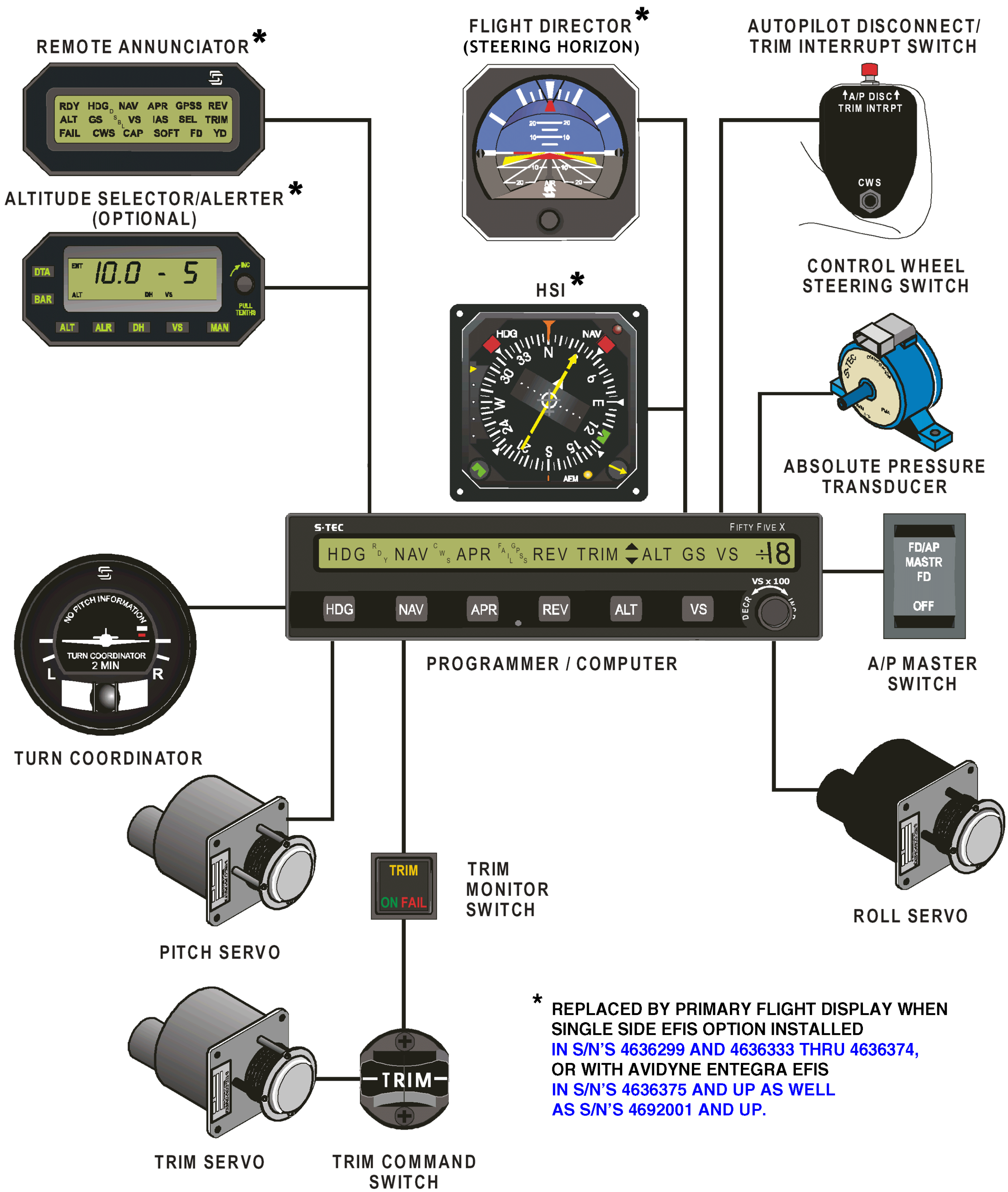
Effectivity
4692001–4692133, 4692141
4692149 and 4692153

S-TEC System 55/55X Installation
Figure 1 (Sheet 5 of 7)



S-TEC System 55/55X Installation
Figure 1 (Sheet 6 of 7)

Effectivity
4692001–4692133, 4692141
4692149 and 4692153



* REPLACED BY PRIMARY FLIGHT DISPLAY WHEN
SINGLE SIDE EFIS OPTION INSTALLED
IN S/N'S 4636299 AND 4636333 THRU 4636374,
OR WITH AVIDYNE ENTEGRA EFIS
IN S/N'S 4636375 AND UP AS WELL
AS S/N'S 4692001 AND UP.

Effectivity
4636160, 4636248–4636459
4636461–4636462, 4636481
4692001–4692133, 4692141
4692149 and 4692153

S-TEC System 55/55X Installation
Figure 1 (Sheet 7 of 7)

C. Description

The S-TEC System 55/55X is a rate-based autopilot that controls the roll and pitch axes of the aircraft. The autopilot's main function is to convert pilot commands to logic signals for the roll and pitch computers. As the pilot enters the desired mode by pressing the appropriate mode selector switch, the computer acknowledges the mode, causing the appropriate annunciator to illuminate.

The Roll Computer receives select input signals from the Directional Gyro (DG) or Horizontal Situation Indicator (HSI), VHF Omnidirectional Radio (VOR), Localizer (LOC) or Global Positioning System (GPS), Deviation Indicators, and the Turn Coordinator. It then computes roll servo commands for stabilization, turns, navigation intercepts, and tracking.

The Pitch Computer receives select input signals from the Altitude Pressure Transducer, Accelerometer, Glideslope Deviation Indicator and Altitude Selector/Alerter (if installed). It then computes pitch servo commands for vertical speed, altitude hold and glideslope intercept and tracking. Sensing for trim annunciation or automatic stabilator trim is provided by the pitch servo. Drive for the stabilator trim servo is provided by the pitch computer.

A typical S-TEC System 55/55X Autopilot installation includes the following:

(1) Panel Mounted:

Programmer/Computer, Turn Coordinator, Annunciator, D.G. or HSI, altitude selector/alerter, and Steering Horizon.

(2) Remote Mounted:

Roll Servo, Pitch Servo, Trim Servo, Trim Monitor, A/P Disconnect switch, and Altitude (Pressure) Transducer.

Servo installations use aluminum brackets to secure the servos to the airframe. Attachment to the airplane's primary flight control and trim systems is accomplished with bridle cables and extension attachments.

D. Troubleshooting

System functionality can be determined using functional checks described in the AFM Supplement and autopilot Pilot's Operating Handbook. More detailed troubleshooting should be accomplished by authorized S-TEC Dealers, holding the appropriate FAA certification, with required test equipment and service data.

E. GPSS (System 55X only.)

The Global Positioning System Steering (GPSS) is a function of the 55X autopilot only. In the GPSS mode, the converter receives ground speed and bank angle digital signals that are calculated and converted to a commanded turn rate. The turn rate is then scaled and converted to a DC heading error signal that is compatible with S-TEC autopilots. The end result is an autopilot that can be directly coupled to the roll steering commands produced by the GPS Navigator, eliminating the need for the pilot to make any further adjustments to the HSI course arrow or the DG's heading bug.

F. System Operation

Operation of the autopilot and other systems is described in the FAA-approved Airplane Flight Manual Supplement (AFMS) - see airplane Pilot's Operating Handbook (POH), Section 9. Specialized controls, annunciation, operation and interpretation are covered in this supplement and in the S-TEC Autopilot POH that supplements the approved AFMS.

G. Maintenance

Except as provided in 5-20-00, servicing and/or maintenance of the autopilot system is On-Condition.

NOTE: Servicing of S-TEC System 55/55X Autopilot installations is best accomplished by approved S-TEC dealers holding the appropriate FAA-certification. Locations of and access to the components installed are described and depicted individually below. Removal and replacement of components is generally indicated by functional checks provided in the AFM Supplement, S-TEC Autopilot POH and/or below.

H. Post-Maintenance Operational Checkout

Complete the following checkout procedure after any maintenance to the system is performed.

NOTE: The Systems 55/55X incorporate a SELF-TEST that requires a 100% pass rate before the autopilot can be engaged.

NOTE: In airplanes equipped with either the optional Meggitt Single Side EFIS (see 34-21-00) or Avidyne Entegra (see 34-22-00), references below to the remote annunciator, flight director and HSI are to those functions in the Primary Flight Display (PFD).

(1) Apply aircraft power.

(2) Avionics Master Switch — ON

(3) Autopilot Master Switch — Set to FD / AP

NOTE: Observe that all segments of the Programmer / Computer display and annunciators illuminate for five (5) seconds during test. Satisfactory completion of the SELF-TEST is indicated when the Ready (RDY) annunciator remains on at the end of the five (5) second self-test. Should a fault be detected, the FAIL annunciator will remain on at the conclusion of the self-test and the autopilot will not operate.

(4) Trim Master (ON / OFF) Switch — ON

(5) HDG and VS switches — PRESS / RELEASE

Ensure that HDG and VS illuminate on the Fifty Five X annunciator.

(6) VS Knob — ROTATE CW

Pitch control (i.e. - the control yoke) should move slowly out (pilot may have to assist a heavy yoke).

(7) VS Knob — ROTATE CCW

Pitch control should move slowly in.

(8) A/P DISC Trim Interrupt Switch (on control yoke) — PRESS

Verify the autopilot disconnects.

(9) HDG Mode — ENGAGE

(10) DG or HSI HDG bug — MOVE LT / RT

Roll control should follow the HDG bug.

NOTE: If HSI equipped, center the course arrow under the lubber line and push the NAV button. Move the course arrow on the HSI left then right. Roll control should follow the course arrow. Channel a valid VOR signal and move course arrow just enough to deflect the left / right needle one (1) or two (2) dots. Roll control should follow the Course Deviation Indicator (CDI) left / right needle during the test. (This test is only valid if the left / right needle is centered with the course arrow under the lubber line.)

NOTE: If DG equipped, center the HDG bug under the lubber line. Channel a valid VOR signal. Move the OBS to cause left / right CDI needle deflection. The roll control should follow the left / right needle movement.

(11) REV Mode button PUSH

Roll control should respond opposite to the course arrow and CDI left / right needle inputs.

(12) Altitude Hold (ALT) button PUSH

Slowly pull out (nose up) on the pitch control (i.e. - control yoke). Autotrim should run nose down with TRIM flashing on the remote annunciator and the autopilot computer / programmer after approximately 3 seconds. Slowly move control yoke forward (nose down). After 3 seconds, autotrim should move nose up with TRIM flashing on the remote annunciator and the autopilot computer / programmer after approximately 3 seconds.

(13) Trim Master (ON / OFF) Switch OFF

(14) Manual Electric Trim Test:

(a) Trim Master (ON / OFF) Switch ON

1) Move each segment of the Manual Electric Trim Command Switch FWD and AFT.

Trim should not run.

2) Move both segments of the Trim Command switch FWD.

Trim should run nose down.

3) Move both segments of the Trim Command switch AFT.

Trim should run nose up.

(b) Re-trim aircraft for takeoff and check controls for freedom of movement. Be sure the autopilot and trim servos are disengaged.

(15) Flight Director Test:

(a) Autopilot Master Switch SELECT FD

Note the roll, pitch and trim servos are disengaged. The steering bar should be in view on the attitude indicator.

(b) HDG Mode ENGAGE

MOVE HDG bug 45 degrees left. The roll steering bar should slowly indicate a left steering command. Repeat the same test for the right side.

(c) VS Mode ENGAGE

SELECT 1500 FPM rate of climb. Note the pitch steering bar moves slowly up. Repeat the same test for the down direction.

(d) Autopilot Master Switch SELECT FD / AP

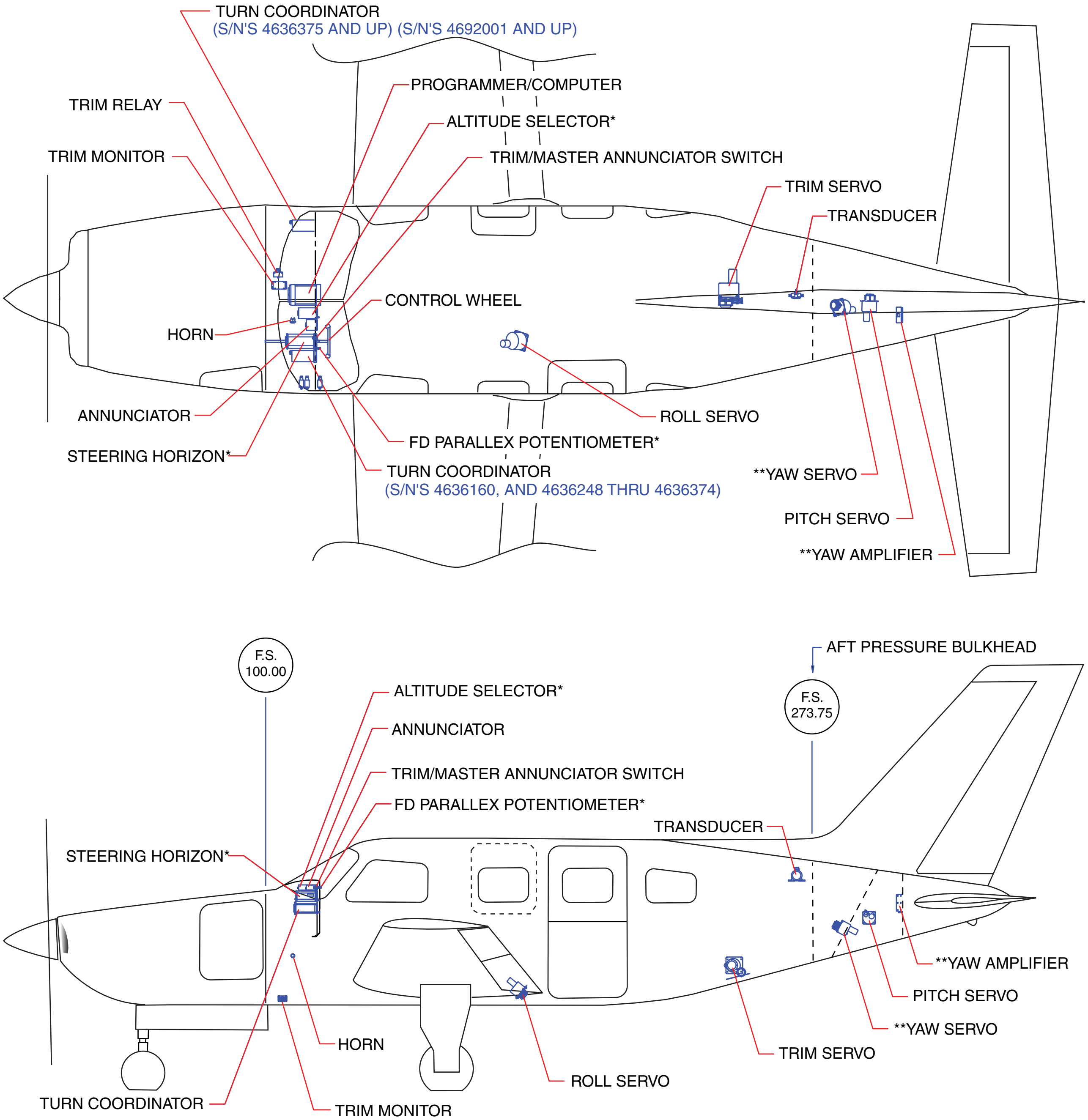
The servos should re-engage.

(e) Trim Master ON / OFF Switch ON

(f) Manual Electric Trim Command Switch MOVE FWD or AFT

The autopilot should disconnect.

NOTE: The Manual Electric Trim Command Switch will disconnect the autopilot only if there is a Pitch Mode engaged.



*COMPONENTS NOT INSTALLED IN S/N'S 4636375 AND UP OR S/N'S 4692001 AND UP.
**COMPONENTS NOT INSTALLED IN S/N'S 4692001 AND UP.

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Component Locator - S-TEC System 55/55X
Figure 2

I. Components

(1) Panel-Mounted Components

The flight director, HSI, autopilot programmer/computer, altitude selector/alerter (if installed), remote annunciator, and turn coordinator are either face-mounted or rack-mounted in the instrument panel. See 39-10-00 for removal and installation instructions.

(2) Component Locator

See “Figure 2” on page 221012.

J. Trim Monitor

The trim monitor is mounted on the underside of the forward-most access panel, in the cockpit floor, at the co-pilot’s feet.

(1) Removal

(a) Remove carpet to expose access panel.

(b) Remove screws holding access panel to floor and open access panel to expose trim monitor and autopilot harness.

(c) Disconnect autopilot harness.

(d) Remove screws (4) holding trim monitor to access panel and remove trim monitor.

(2) Installation

(a) Place trim monitor in position on underside of access panel and secure with screws (4).

(b) Connect autopilot harness.

(c) Place access panel in position and secure with screws.

(d) Replace carpet over access panel.

(e) Check system operation - perform Manual Electric Trim Test under Post-Maintenance Operational Checkout, above.

K. Roll Servo

See “Figure 3” on page 221015.

The roll servo is mounted underneath the left aft-facing passenger seat or the entertainment cabinet, whichever is installed.

(1) Removal

(a) Remove the aft-facing passenger seat or entertainment cabinet, whichever is installed.

(b) Remove adjacent carpet.

(c) Remove screws securing centerline floor access panel and remove access panel.

(d) Disconnect autopilot harness.

(e) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from aileron balance and autopilot bridle cables.

(f) Remove nuts, washers, and bolts (4 ea.) securing roll servo to mounting brackets and remove roll servo with attached bridle cable.

(2) Installation

(a) Rig ailerons per Aileron Control Rigging and Adjustment, 27-10-00.

(b) Lock the control wheels together in the aileron neutral position using a suitable tool.

(c) Lock the ailerons in neutral position using a suitable contour fixture at the inboard ends of the ailerons. Neutral position is defined by wing loft contour at this location.

(d) Remove screws (4) and remove capstan cover and cable guards from servo.

(e) Adjust roll servo clutch torque per Servo Clutch Torque Adjustment, below.

(f) Wrap autopilot bridle cable, align capstan, and and tighten center-ball setscrew as shown in Figure 3.

(g) Replace cable guards and capstan cover, secure with screws (4).

(h) Position servo as shown in Figure 3 and install and secure nuts, washers, and bolts (4 ea.) holding servo to mounting brackets.

(i) Position cable clamps (2) as shown in “Figure 3” and tighten nuts and bolts (2 ea.). Adjust cable clamps in or out along the aileron balance cable to obtain a bridle cable tension as shown in “Figure 3”. Torque cable clamp bolts as specified in “Figure 3”.

(j) Remove the locking fixtures. Aileron neutral position should be maintained at 0 ± 1 degree with the control wheels in neutral.

(k) Use the pilot’s control wheel to move the ailerons and verify aileron travel at the primary stops (at the aileron drive sectors on the wing spars) remains 18 ± 1 degree up and down.

(l) Use the pilot’s control wheel to hold the ailerons firmly against the primary stops. Verify the secondary aileron stops (on the forward pressure bulkhead) retain a cushion (gap) of 0.09 ± 0.03 inch in each direction.

(m) If either step (i) or (j) reveal the aileron controls out of rig, repeat steps (a) thru (j), above.

(n) Connect autopilot harness.

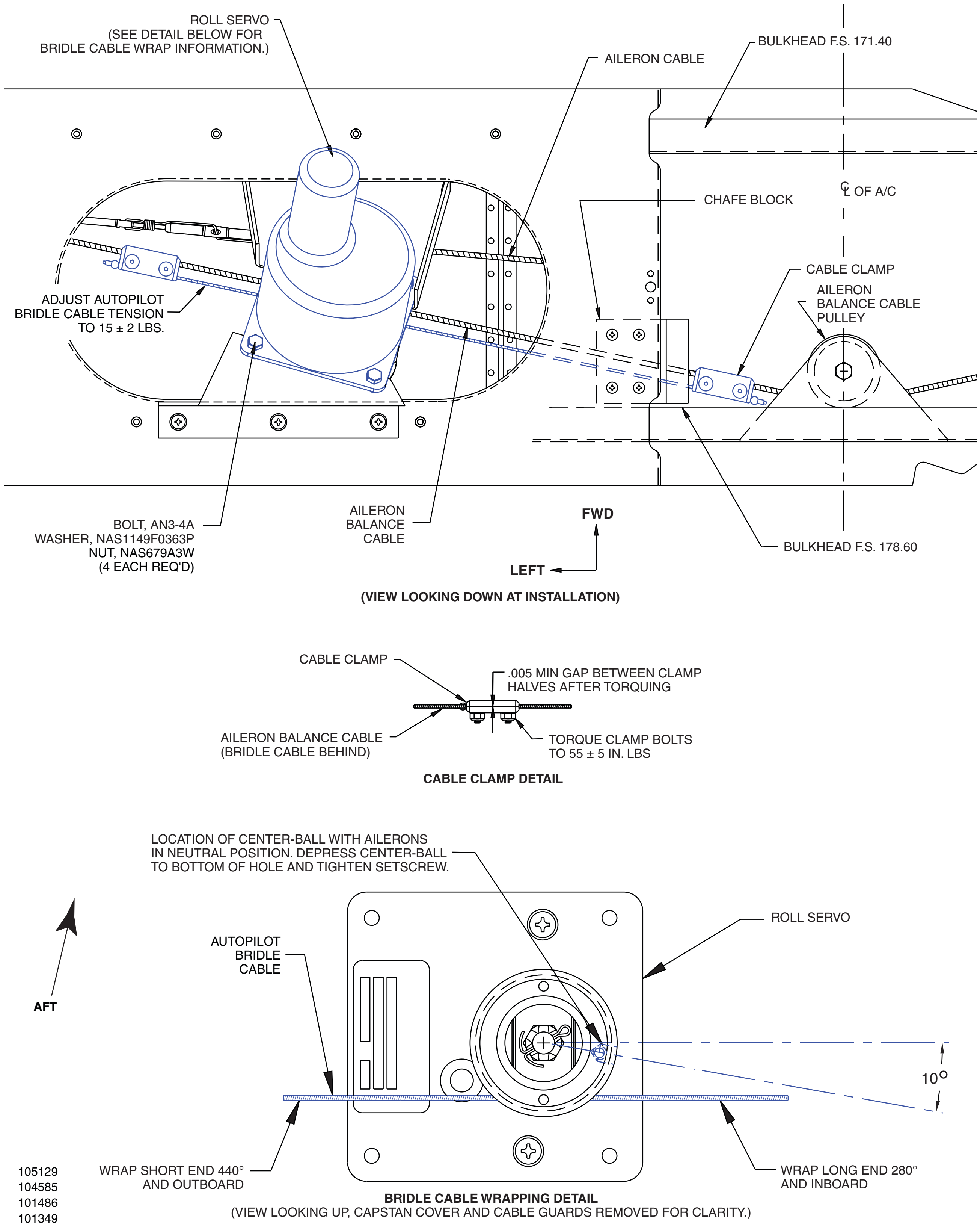
(o) Check autopilot system for binding and proper operation, see “Post-Maintenance Operational Checkout” on page 221010.

(p) Check aileron controls for free and correct movement.

(q) Replace centerline floor access panel and secure with screws.

(r) Replace carpeting.

(s) Replace the aft facing passenger seat or entertainment cabinet, whichever is installed.



Roll Servo Installation
Figure 3

Effectivity
System 55X shown
System 55 similar

L. Pressure Transducer

The pressure transducer is mounted just forward of the aft pressure bulkhead (F.S. 273.75) slightly to the right of the aircraft centerline and just below the aircraft upper skin. Access is through the cabin rear closeout panel.

(1) Removal

(a) Disconnect the transducer from the static system by cutting the Ty-wrap and pulling the static hose off of the transducer hose barb.

(b) Disconnect the autopilot harness.

(c) Remove screws, washers, and nuts (2 ea.) and remove transducer.

(2) Installation

(a) Place transducer in position on bracket. Secure to bracket with screws, washers, and nuts (2 ea.)

(b) Slide the static hose over the transducer hose barb and secure with a Ty-wrap.

(c) Connect the autopilot harness.

M. Trim Servo

See “Figure 4” on page 221017.

The trim servo is located on the centerline underneath the baggage compartment floor, just forward of F.S. 249.60.

(1) Removal

(a) Remove rear seats. Remove carpet and access panels from baggage compartment floor.

(b) Remove the centerline access panels in the cabin floor to expose the plastic tube that protects the trim cables.

(c) Remove the pins from both ends of the plastic tube.

(d) Slide the tube forward.

(e) Rotate the elevator trim control wheel until the right trim cable turnbuckle clears the aft end of the tube.

(f) Tie a pull rope to the right trim cable forward of the exposed turnbuckle.

(g) Tie a pull rope to the right trim cable aft of the idler pulley.

(h) Tie-off both pull ropes to maintain tension on the left trim cable and the extreme forward and aft segments of the right trim cable (i.e., forward of the turnbuckle and aft of the idler pulley).

NOTE: Tension must be applied to the trim cables to prevent the cables from unwrapping from the trim wheel drum or the trim actuator, and to prevent the cables from fouling at any of the pulleys.

(i) Slack off the exposed turnbuckle to relieve tension on the right trim cable segment between the turnbuckle and idler pulley.

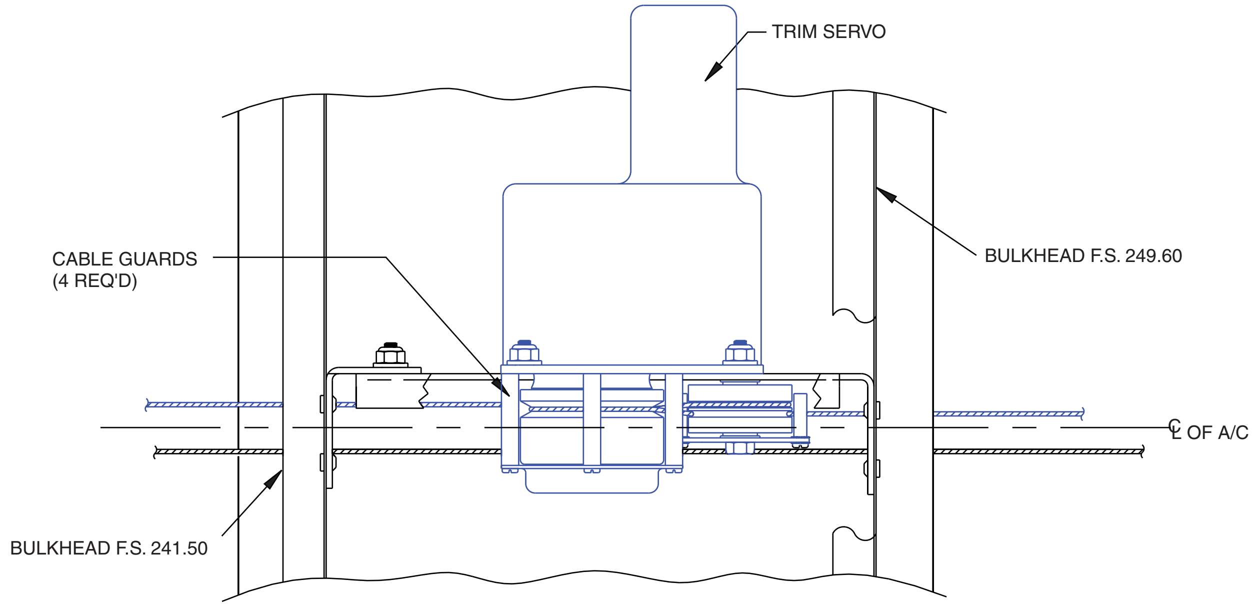
(j) Disconnect the autopilot harness.

(k) Remove the capstan cover and cable guards (4) by removing the retaining screws (4).

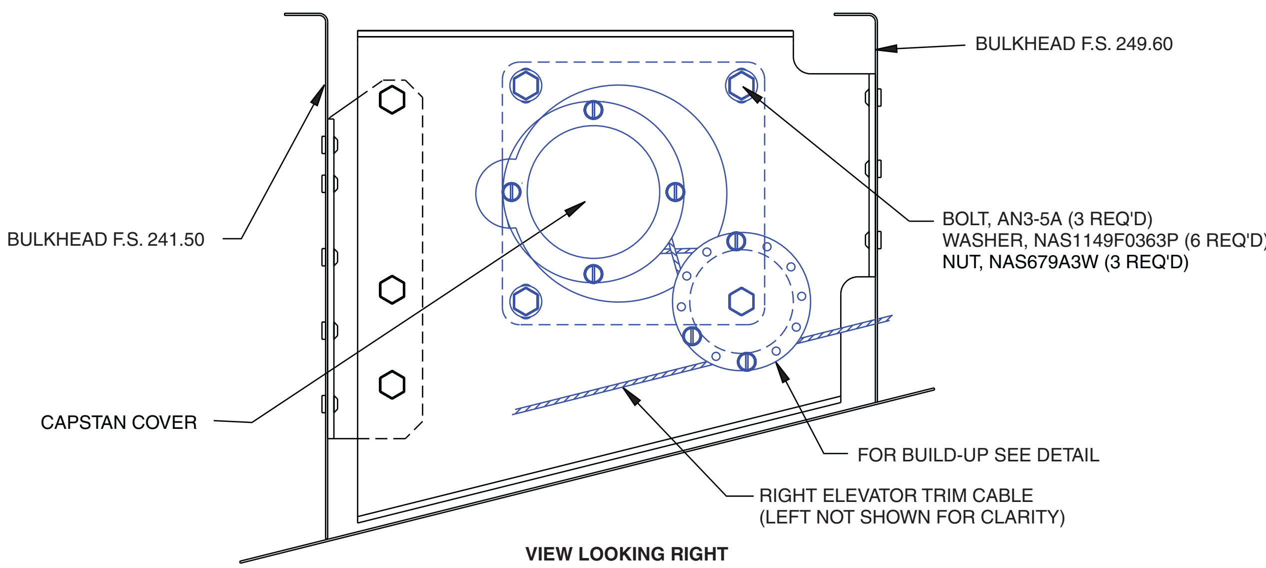
(l) Remove the bolt, nut, and washer securing the idler pulley to the trim servo baseplate and mounting bracket and remove the idler pulley components.

NOTE: The idler pulley breaks down into the following components upon removal of the bolt, above: mounting plate/cable guard assembly, idler pulley, and two washers.

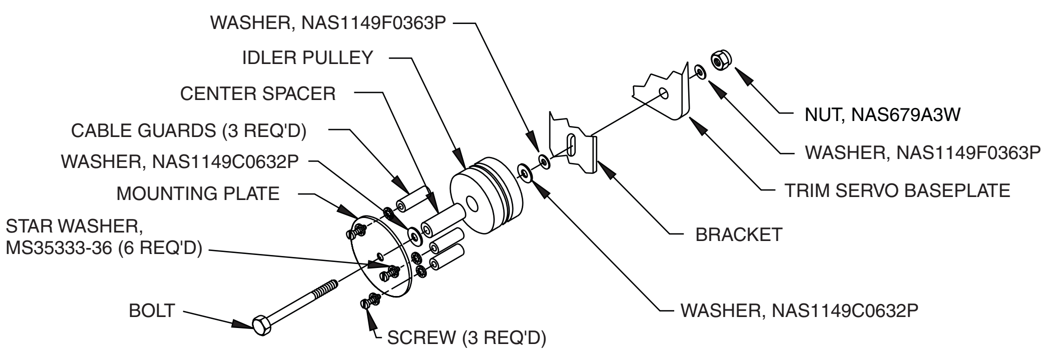
(m) Remove the remaining bolts, nuts, and washers (3 ea.) securing the trim servo to its mounting bracket and remove the trim servo.



VIEW LOOKING DOWN AT TRIM SERVO INSTALLATION



VIEW LOOKING RIGHT



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IDLER PULLEY DETAIL

Trim Servo Installation
Figure 4

Effectivity
System 55X shown
System 55 similar

(2) Installation

(a) Adjust trim servo clutch torque per Servo Clutch Torque Adjustment, below.

(b) With the capstan cover and cable guards removed, position the trim servo as shown in "Figure 4" on page 221017. Secure with bolts, nuts, and washers (3 ea. - i.e., top two and bottom forward).

(c) Assemble the idler pulley cable guards (3) to the mounting plate with screws (1 ea.) and star washers (2 ea.). Place the center bolt through the mounting plate/cable guard assembly and slide a washer over the threaded end and up against the mounting plate. Set the mounting plate/cable guard/bolt assembly aside.

(d) Drape the slack left trim cable over the servo capstan.

(e) Place thumb and forefinger on top of the capstan over the trim cable in its groove. Pressing the trim cable into its groove, slide thumb and forefinger down around opposite sides of the servo capstan and pull the trim cable slack to the lower rear quadrant.

(f) Holding the trim cable in that position, install the capstan cover and cable guards as shown in "Figure 4".

(g) Hold the idler pulley on the forward side of the trim servo and align it with the left trim cable. Move the idler pulley down to the left trim cable forward of the trim servo and capture the trim cable in the left cable groove on the bottom of the idler pulley.

(h) Slide the idler pulley along the trim cable, underneath the servo capstan, and bring it approximately to its installed position (see "Figure 4"). At this point, the forward portion of the trim cable should be routed under, around the rear, and over the top of the idler pulley and under, around the front, and over the top of the capstan.

(i) Holding the idler pulley in this position, reach to the rear and pull the aft portion of the trim cable under the idler pulley and seat it in the right cable groove on the idler pulley. The left trim cable should now be routed as shown in "Figure 4".

(j) Place the spacer inside the idler pulley.

(k) Position the mounting plate/cable guard/bolt assembly as shown in Figure 4 and slide the bolt through the spacer inside the idler pulley.

(l) Place a washer over the bolt end and put the bolt through the trim servo mounting bracket and baseplate. Secure with a nut and washer, taking care to ensure that the cable guards are positioned, and the left trim cable is routed, as shown in "Figure 4".

(m) Rig elevator trim per Elevator Trim Controls Rigging, 27-30-00.

(n) Connect the autopilot harness.

(o) Check autopilot system for proper operation.

(p) Check elevator trim controls for free and correct movement.

(q) Replace access panels and carpet in cabin and baggage compartment floors.

(r) Replace rear seats.

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N. Yaw Servo (PA-46-350P only.)

See “Figure 5” on page 221021.

The yaw servo is located in the unpressurized aft fuselage.

(1) Removal

(a) Remove the tailcone access panel (see #1, Figure 3, 6-00-00).

(b) Disconnect autopilot harness.

(c) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from rudder cables and autopilot bridle cables.

(d) Remove nuts, washers, and bolts (4 ea.) securing yaw servo to mounting brackets and remove yaw servo with attached bridle cable.

(2) Installation

(a) Rig rudder controls per Rudder Control Rigging, 27-20-00.

(b) Lock rudder in its neutral (streamlined) position.

(c) Remove screws (4) and remove capstan cover and cable guards from servo.

(d) Adjust yaw servo clutch torque per Servo Clutch Torque Adjustment, below.

(e) Wrap autopilot bridle cable, align capstan, and and tighten center-ball setscrew as shown in “Figure 5”.

(f) Replace cable guards and capstan cover, secure with screws (4).

(g) Position yaw servo as shown in “Figure 5” and secure with bolts, nuts, and washers (4 ea.)

(h) Remove tape or clip holding bridle cable on capstan and attach left and right autopilot bridle cables to the left and right rudder cables with cable clamps (2).

(i) Evenly adjust the cable clamps up the rudder cables until the tension on both left and right autopilot bridle cables is as specified in “Figure 5”.

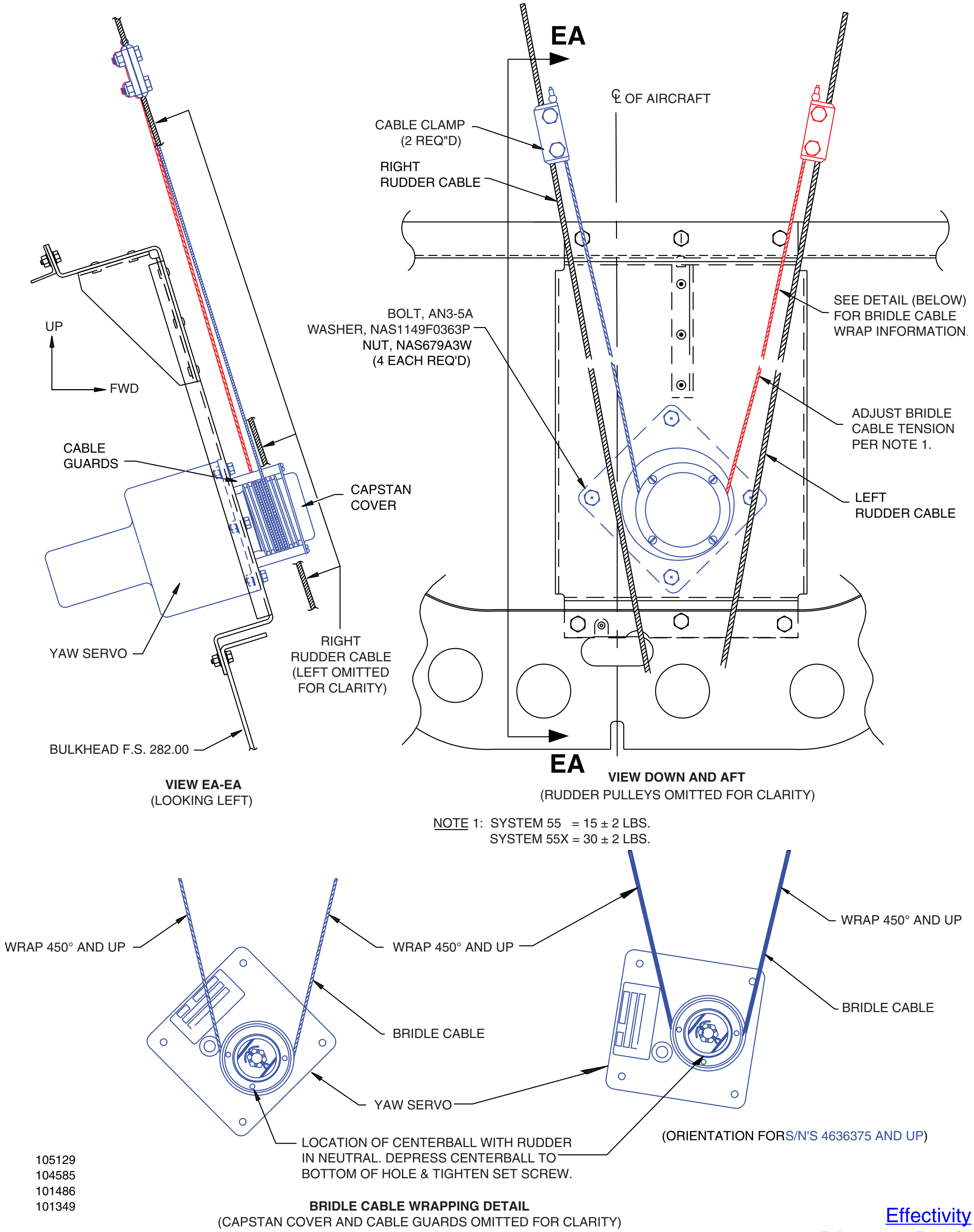
(j) Connect autopilot harness.

(k) Remove rudder lock.

(l) Check autopilot system for binding and correct operation.

(m) Check rudder controls for free and correct movement.

(n) Replace and secure the tailcone access panel.



Yaw Servo Installation
Figure 5

Effectivity
PA-46-350P only
System 55X shown
System 55 as noted

O. Pitch Servo

See “Figure 6” on page 221023.

The pitch servo is located in the unpressurized aft fuselage.

(1) Removal

(a) Remove the tailcone access panel (see #1, Figure 3, 6-00-00).

(b) Disconnect autopilot harness.

(c) Remove locking clips (2) from turnbuckle in autopilot bridle cable. Let the upper segment of the autopilot bridle cable remain on top of the autopilot sector assembly.

(d) Remove cotter pins (cable guards) (2) holding the lower segment of the autopilot bridle cable on idler pulley and remove the bridle cable from the idler pulley.

(e) Remove cotter pin (cable guard) holding the autopilot bridle cable to the bottom of the autopilot sector assembly.

(f) Remove cotter pin, clevis pin, washers (2), and autopilot bridle cable fork end from autopilot sector assembly.

(g) Remove nuts, washers, and bolts (4 ea.) securing pitch servo to mounting brackets and remove pitch servo with attached bridle cable.

(2) Installation

(a) Rig elevator controls per Elevator Controls Rigging, 27-30-00.

(b) Remove screws (4) and remove capstan cover and cable guards from servo.

(c) Adjust pitch servo clutch torque per Servo Clutch Torque Adjustment, below.

(d) Wrap autopilot bridle cable, align capstan, and and tighten center-ball setscrew as shown in “Figure 6”.

(e) Replace cable guards and capstan cover, secure with screws (4).

(f) Position pitch servo as shown in “Figure 6” and secure with bolts, nuts, and washers (4 ea.).

(g) Position autopilot bridle cable fork end on autopilot sector assembly and secure with clevis pin, washers (2), and cotter pin.

(h) Ensuring bridle cable is in bottom groove of sector assembly, insert and secure cotter pin (cable guard).

(i) Loop bridle cable over idler pulley and install cotter pins (cable guards) (2).

(j) Connect bridle cable turnbuckle and adjust cable tension as specified in “Figure 6”.

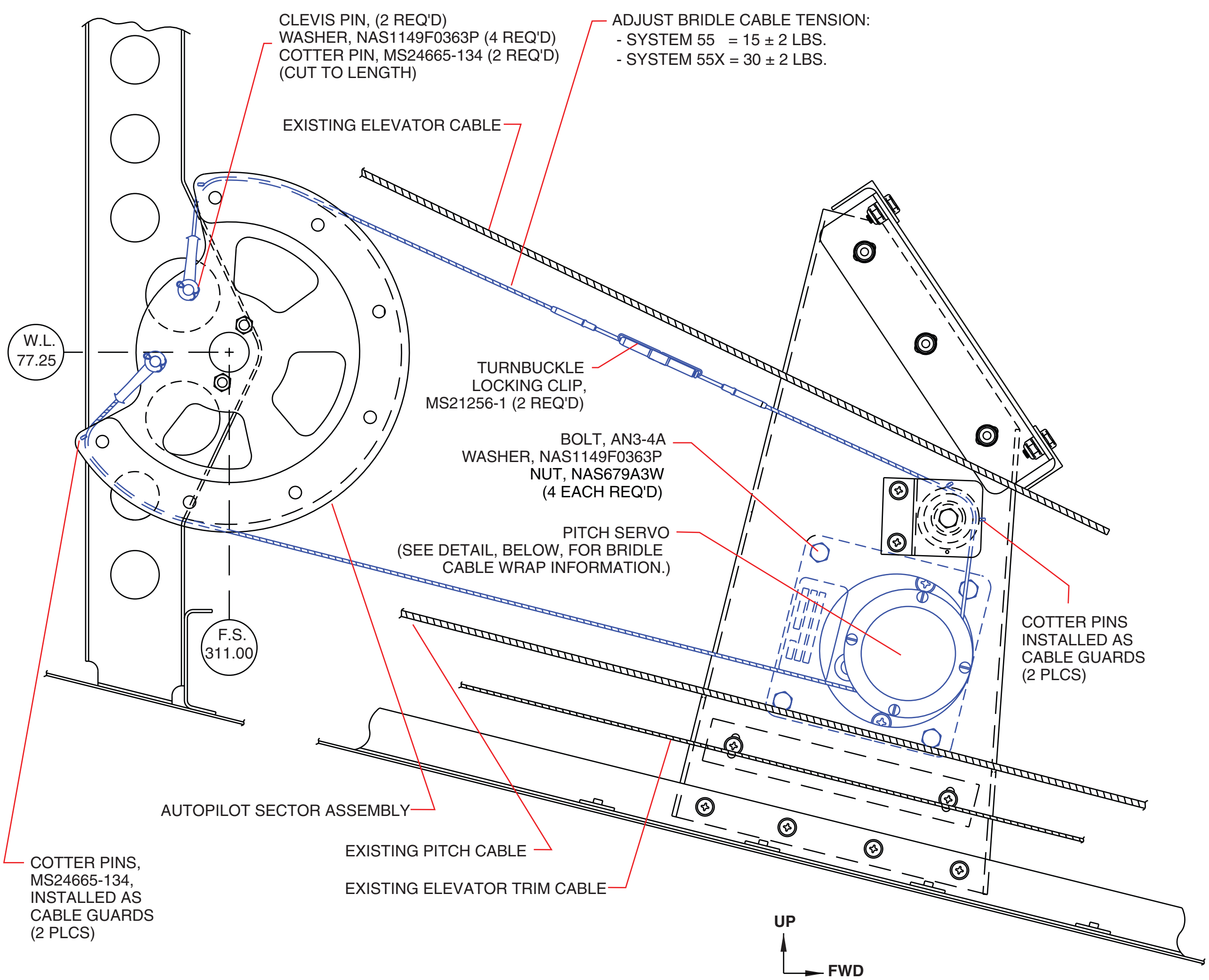
(k) Install bridle cable turnbuckle locking clips (2).

(l) Connect autopilot harness.

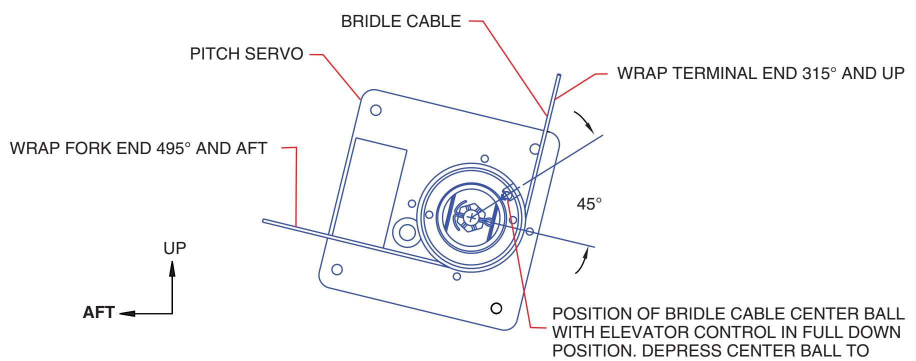
(m) Check autopilot system for binding and correct operation.

(n) Check elevator controls for free and correct movement.

(o) Replace and secure the tailcone access panel.



VIEW LOOKING LEFT FROM CENTERLINE OF AIRCRAFT
(ELEVATOR SECTOR OMITTED FOR CLARITY)

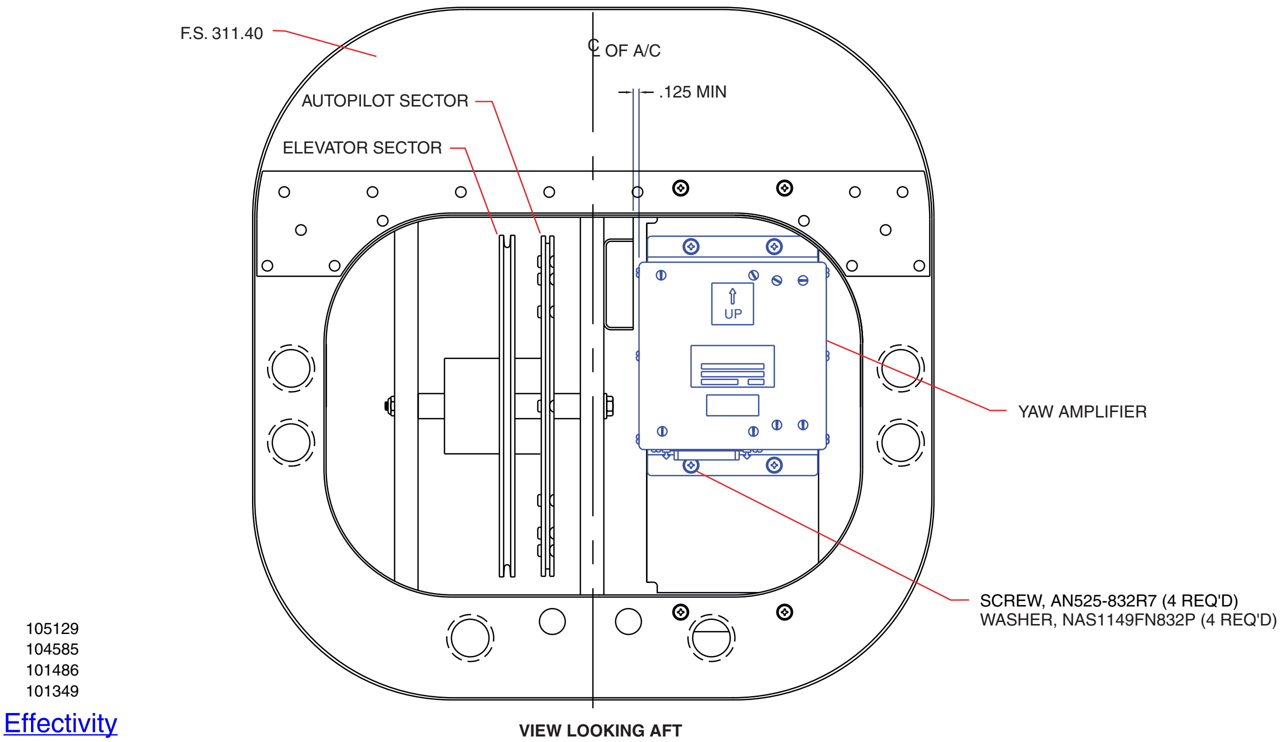


105129
104585
101486
101349

BRIDLE CABLE WRAPPING DETAIL
(CAPSTAN COVER AND CABLE GUARDS OMITTED FOR CLARITY)

Pitch Servo Installation
Figure 6

Effectivity
System 55X shown
System 55 as noted



Effectivity
PA-46-350P only
System 55X shown
System 55 similar

Yaw Amplifier Installation
Figure 7

P. Yaw Amplifier (PA-46-350P only.)

See "Figure 7" on page 221024.

The yaw amplifier is located in the unpressurized aft fuselage.

(1) Removal

(a) Remove the tailcone access panel (see #1, Figure 3, 6-00-00).

(b) Disconnect autopilot harness.

(c) Remove screws and washers (4 ea.) and remove the yaw amplifier.

(2) Installation

(a) Position yaw amplifier and secure with screws and washers (4 ea.)

(b) Connect autopilot harness.

(c) Check autopilot system for correct operation.

(d) Replace and secure the tailcone access panel.

(3) Post-Installation Ground Checks

(a) Level aircraft laterally per 8-20-00.

(b) Center trim pot on instrument panel.

(c) Align the cap on the yaw trim pot knob (on instrument panel) so that the white line is centered vertically.

(d) Adjust the pot in the yaw amplifier (located under plug in top of amplifier) to provide a null (i.e., no servo action) at the yaw servo.

(e) Align the cap on the yaw trim pot knob (on instrument panel) so that the white line is centered vertically.

(f) Check operation of yaw trim pot by turning trim pot both left and right and verify rudder pedal movement in the same direction (i.e., turning trim pot left causes left rudder pedal depression, etc.)

Q. Servo Clutch Torque Adjustment

See "Figure 8" on page 221026.

(1) Remove servo per instructions under specific servo, above.

(2) Place servo in a holding fixture (i.e., vice) with capstan up.

(3) Remove capstan cover, cable guards, and cable.

(4) Check capstan torque by attaching the capstan adjusting tool (special tool - see parts catalog) to the capstan and using a currently calibrated torque wrench as shown in "Figure 8".

(a) Acceptable torque is specified in "Chart 1".

(b) If adjustment is required, proceed as follows.

(5) Remove cotter pin from end of servo shaft and remove castle nut, shim washers, and tension washers.

(6) Replace tension washers as required (see "Chart 1").

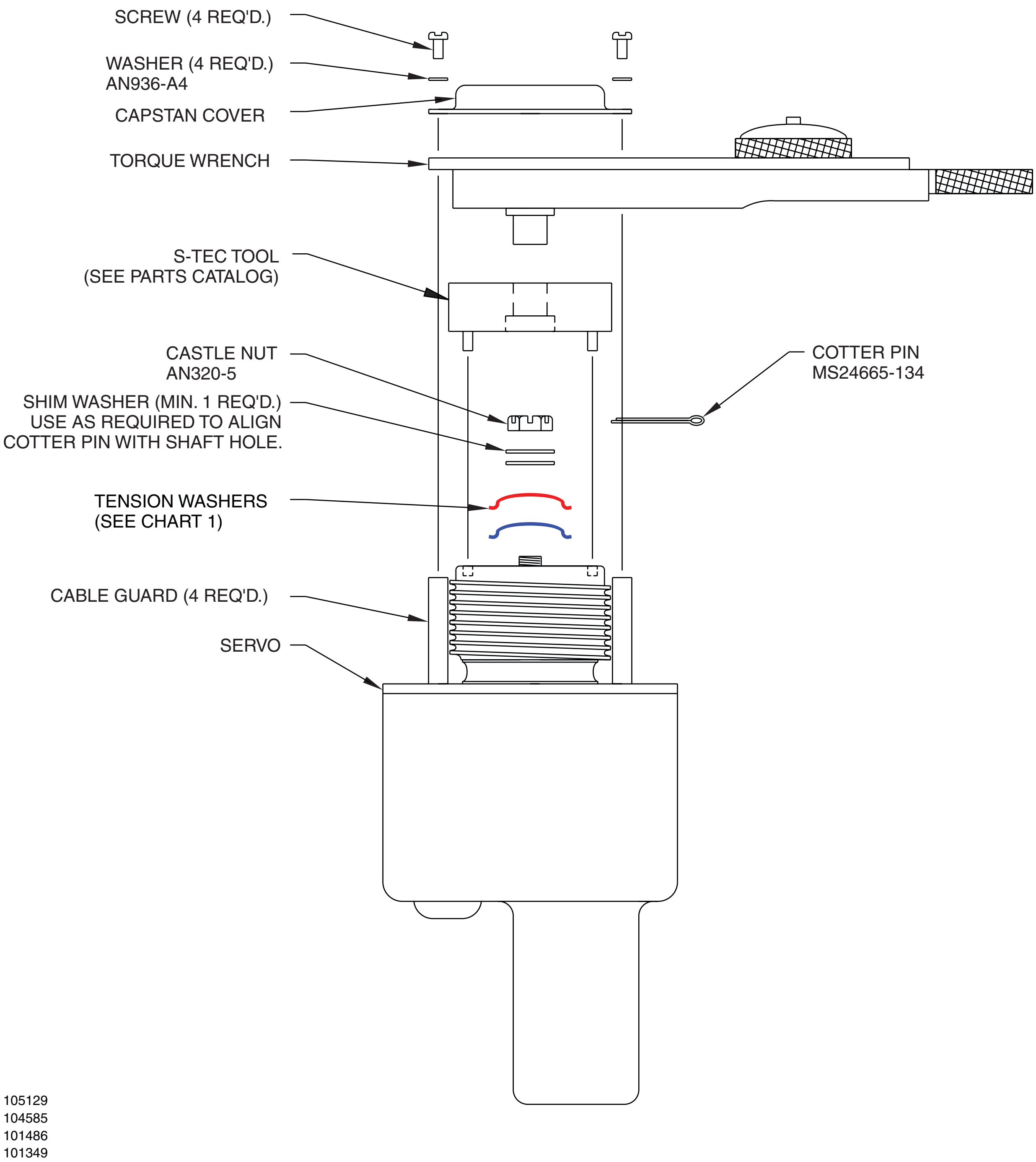
(7) Replace shim washers and castle nut.

(8) Tension castle nut so that capstan torque is as specified in "Chart 1".

CHART 1
S-TEC SYSTEM 55/55X AUTOPILOT SERVO CLUTCH TORQUE

SERVO	TORQUE (In. Lbs.) System 55/55X	WASHERS System 55/55X
Roll	40 ± 2 (1)	Three P/N 1253-1 (Blue)
Pitch (P/N 0108-P4 (2))	27 ± 2 (1)	One P/N 1253 (Red) Two P/N 1253-1 (Blue)
Pitch (P/N 0108-15-P4 (3))	33 ± 3	Three P/N 1253 (Red)
Trim	21 ± 2 (1)	Three P/N 1253 (Red)
Yaw	30 ± 2 (1)	One P/N 1253 (Red) Two P/N 1253-1 (Blue)

(1) ± 3 in S/N's 4636375 and up and in S/N's 4692001 and up.
(2) S/N's 4636160, 4636248 thru 4636441; 4692001 thru 4692021, less 4692017.
(3) S/N's 4636442 and up; 4692017, 4692022 and up.



Effectivity
System 55X shown
System 55 similar

Servo Clutch Torque Adjustment
Figure 8

3. Garmin GFC 700

This system was adopted in 2010 and is installed as standard equipment in S/N's 4636460, 4636463 and up; and S/N's 4692134 and up.

A. Description

See "Figure 9" on page 221028 and "Figure 11" on page 221040, also see 34-25-01 (G1000) or 34-25-02 (G1000 NXi).

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY.

CAUTION: THIS SECTION PROVIDES INFORMATION SUFFICIENT ONLY TO LOCATE, REMOVE, AND INSTALL COMPONENTS. DETAILED TROUBLESHOOTING OR REPAIR IS REQUIRED, IT SHOULD BE PERFORMED BY A QUALIFIED AVIONICS REPAIR SHOP USING THE APPROPRIATE MAINTENANCE MANUALS.

The GFC 700 Automatic Flight Control System (AFCS) provides a three-axis fail-safe digital flight control with Flight Director, Autopilot, Pitch Auto-Trim and Yaw Damper functions. The GFC 700 functionality is distributed across several units of the Garmin G1000/G1000 NXi avionics system and is designed for precision approach to Category 1 minimums. The controls are located on the GMC 710 Mode Controller bezel. The mode logic and Flight Director computations are performed within the GIA 63W Integrated Avionics units. The autopilot computations and monitoring are performed within the GSA 8X Servo Actuators with GSM 85A Servo Mounts. Dual Control Display Units and Integrated Avionics units provide redundancy.

The GFC 700 AFCS is mainly controlled through the GMC 710 AFCS Control Unit. See "Figure 9". See also 34-25-01 (G1000) or 34-25-02 (G1000 NXi), as appropriate:

The AP DISC (Autopilot Disconnect) Switch, CWS (Control Wheel Steering) Button, GO AROUND Switch, and MEPT (Manual Electric Pitch Trim) Switch are additional AFCS controls and are located in the cockpit, separately from the AFCS Control Unit.

B. Troubleshooting

See the following under Integrated Avionics System (IAS) in 34-25-01 (G1000) or 34-25-02 (G1000 NXi), as appropriate:

(1) Troubleshooting

(2) LRU Test Procedures

(a) GCU 476 Test

(b) GMC 710 Test

C. Ground Checks

(PIR-PPS55026, Rev. F; 107977 F; 107998 F.)

(1) Setup

(a) Ensure G1000/G1000 NXi software is loaded, properly configured, and checked out per 34-25-01 (G1000) or 34-25-02 (G1000 NXi), as appropriate:

(b) Level the airplane per 8-20-00.

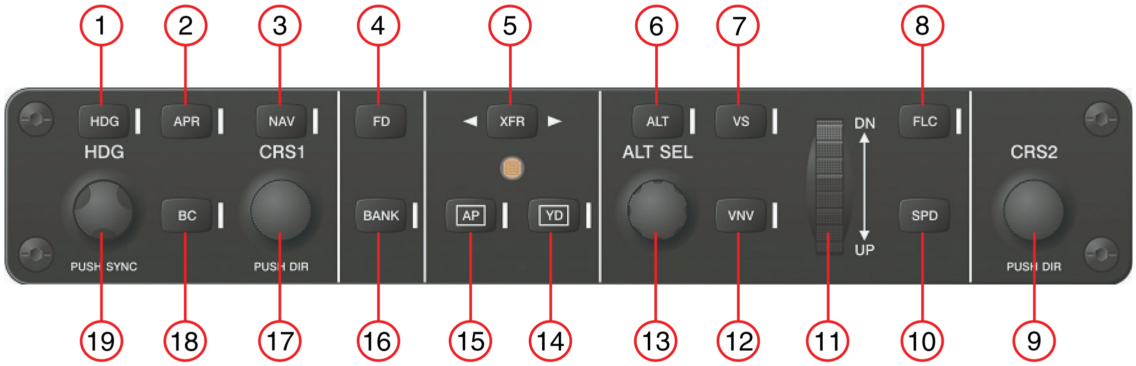
(c) Apply external power.

(2) References

(a) The appropriate Garmin G1000/G1000 NXi Cockpit Reference Guide.

(b) Appropriate Pilot's Operating Handbook (POH) serialized to the airplane.

(c) Avionics/Electrical System Installation schematics found in Chapter 91.



AFCS Control Unit (GMC 710)

1	**HDG Key**	Selects/deselects Heading Select Mode.
2	**APR Key**	Selects/deselects Approach Mode.
3	**NAV Key**	Selects/deselects Navigation Mode.
4	**FD Key**	Activates/deactivates the flight director in the default pitch and roll modes. If the autopilot is engaged, the FD Key is disabled.
5	**XFR Key**	Switches the autopilot between the pilot-side and the copilot-side flight directors. This selection also selects which air data computer is communicating with the active transponder and which PFD triggers the altitude alert. Upon power-up, the pilot-side FD is selected.
6	**ALT Key**	Selects/deselects Altitude Hold Mode.
7	**VS Key**	Selects/deselects Vertical Speed Mode.
8	**FLC Key**	Selects/deselects Flight Level Change Mode.
9	**CRS2 Knob**	Sets the copilot-selected course on the HSI of PFD2 when the VOR1, VOR2, or OBS/SUSP mode is selected. Pressing this knob centers the CDI on the currently selected VOR. The copilot-selected course provides course reference to the copilot-side flight director when operating in Navigation and Approach modes.
10	**SPD Key**	Has no function.
11	**NOSE UP/DN Wheel**	Controls the active mode reference for the Pitch, Vertical Speed, and Flight Level Change modes.
12	**VNV Key**	Selects/deselects Vertical Navigation mode.
13	**ALT SEL Knob**	Sets the selected altitude in the Selected Altitude Box. In addition to providing the standard G1000/G1000 NXi altitude alerter function, selected altitude provides an altitude setting for the Altitude Capture/Hold mode of the AFCS.
14	**YD Key**	Engages/disengages the yaw damper.
15	**AP Key**	Engages/disengages the autopilot.
16	**BANK Key**	Selects/deselects Low Bank Mode.
17	**CRS1 Knob**	Sets the pilot-selected course on the HSI of PFD1 when the VOR1, VOR2, or OBS/SUSP mode is selected. Pressing this knob centers the CDI on the currently selected VOR. The pilot-selected course provides course reference to the pilot-side flight director when operating in Navigation and Approach modes.
18	**BC Key**	Selects/deselects Back Course Mode.
19	**HDG Knob**	Sets the selected heading on the HSI. When operating in Heading Select mode, this knob provides the heading reference to the flight director.

GMC 710 AFCS Control Unit
Figure 9

NOTE: In the sections that follow, an Autopilot disconnect should be accompanied by an aural alert (two-second tone) unless otherwise specified.

NOTE: Items bracketed by double daggers ‡ ‡ in the following AFCS procedures apply to PA-46-350P S/N's 4636460, 4636463–4636632, 4636634–4636651; and PA-46-350T S/N's 4692134–4692140, 4692141–4692148, 4692150–4692152, 4692154–4692214 only

NOTE: Items in square brackets [ ] in the following AFCS procedures apply to PA-46-350P S/N's 4636633, 4636652 and up.

NOTE: Items in curly brackets { } in the following AFCS procedures apply to PA-46-350P S/N's 4636716, 4636720 and up.

(3) Pre-Flight Test (PFT)

(a) Set the AV BUS MASTR switch to OFF. After 30 seconds select the AV BUS MASTR switch to ON. Verify the GFC 700 begins an automatic pre-flight test after AHRS and ADC parameters become valid.

(b) Verify that a white 'PFT' annunciation is displayed on PFD1 and PFD2. A momentarily red 'AFCS' annunciation displayed before PFT starts is acceptable.

(c) Upon successful completion of the test, an aural alert will sound and the annunciation will clear. If the 'PFT' annunciation turns red, the test has failed. Refer to Troubleshooting under Integrated Avionics System in 34-25-01 (G1000) or 34-25-02 (G1000 NXi), as appropriate.

(d) Repeat Steps (a)–(c) to test the PFT aural alert for the other GIA. (For even-interval system power-up, the aural alert is generated by GIA #1, whereas odd-interval system power-ups are generated by GIA #2.)

(4) AFCS Switch Checks

Verify that the AFCS system buttons and switches are operating correctly by performing the following steps (for both the pilot and copilot control yokes where applicable).

(a) Actuate both sections of the PITCH TRIM (TRIM UP/TRIM DN) switch to activate Manual Electric Pitch Trim (MEPT). Verify the trim clutch engages the trim wheel and trim tab drive in the requested direction, and the trim indicator lights within 2–3 seconds [TRIM is annunciated on the PFDs]. Check operation in both the up and down direction.

(b) Activate Copilot's MEPT switch in the UP/DN direction. Then activate the pilot's MEPT switch in the opposite direction. Verify the pilot's MEPT switch takes priority.

(c) Activate each half of the MEPT switch. Verify the trim clutch does not engage, the trim wheel does not move, and PTRM annunciation on the PFD after approximately 3 seconds.

(d) Press the AP DISC TRIM INTER switch and hold while actuating the manual electric trim switch. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.

(e) Activate the MEPT switch. Remove power from the Pitch Trim servo by pulling the PITCH TRIM circuit breaker. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.

(f) Pull AUTO PILOT circuit breaker. Reset PITCH TRIM and AUTO PILOT circuit breakers.

(g) Engage the autopilot by pressing the AP key on the AFCS mode controller (GMC 710). Verify the pitch and roll clutches engage and resist movement of the control wheel. Press and hold the CWS switch and verify the control wheel moves freely when moved manually.

(h) Release the CWS switch and press the AP DISC TRIM INTER switch on the pilots control wheel. Verify the autopilot disengages with a flashing amber 'AP' annunciation on PFD1 and PFD2, accompanied by an aural alert. Verify that the control wheel is free in pitch and roll axes.

(i) Press the GO AROUND [TO/GA] button on the power lever. Verify “GA” [“TO”] is annunciated on PFD1 and PFD2 for both PITCH and ROLL modes and the command bars should be at 8 [7] degrees nose up and wings-level.

(j) Press the Flight Director (FD) key on the AFCS mode controller to deactivate the GA mode. Press the AP key to engage the autopilot. Press the CWS button for a minimum of 5 seconds and release; verifying there is no residual force on the control stick for the pitch and roll axis while the CWS is pressed. Servos reengage after CWS is released.

(k) Disengage the autopilot by pressing the AP DISC TRIM INTER switch on the Copilot’s control wheel. Engage VS mode by pressing the VS key on the AFCS mode controller. Verify PFD1 and PFD2 display ‘VS’ in green and indicates a pitch reference of ‘0 FPM’.

NOTE: The Alt. Sel must be activated for the FLC key to function properly.

(l) Press the FLC key on the AFCS mode controller and verify that ‘FLC’ is annunciated on PFD1 and PFD2 in green with a reference of 78 KTS (minimum airspeed reference for the PA-46-350P Mirage & PA-46R-350T Matrix).

(m) Press the ALT key on the AFCS mode controller and verify that the ‘ALT’ annunciation is displayed in green on PFD1 and PFD2 with an altitude reference equal to the aircraft altitude (within the nearest 20 feet [10 feet]).

(n) Press the FD key and verify that the mode annunciations and command bars are removed from the display.

(5) Autopilot Disconnect Checks – Normal Mode

Engage the autopilot by pressing the AP key on AFCS mode controller (GMC 710) prior to each of the following checks and reset the circuit breaker (if required) prior to the subsequent check. For each check, verify a five-second flashing amber AP annunciation, a three second disconnect aural alert, and FD bars remaining on display.

(a) Press the MEPT switch on the pilot’s control yoke.

(b) Press and hold the left section of the MEPT switch on the pilot’s control yoke. Verify the trim wheel rotates freely when moved manually.

(c) Press and hold the right section of the MEPT switch on the pilot’s control yoke. Verify the trim wheel rotates freely when moved manually.

(d) [Press the MEPT switch on the co-pilot’s control yoke.]

(e) Press and hold the left section of the MEPT switch on the co-pilot’s control yoke. Verify the trim wheel rotates freely when moved manually.

(f) Press and hold the right section of the MEPT switch on the co-pilot’s control yoke. Verify the trim wheel rotates freely when moved manually.

(g) Press GA [TO/GA] button on the throttle handle. (Press the FD key to remove the GA mode annunciation from the display).

NOTE: If enhanced AFCS option is installed, this will not disconnect the autopilot.

(h) [Press the AP button on the GMC.]

(6) Autopilot and Yaw Damper Disconnect Checks – Normal Mode

Engage the autopilot by pressing the AP key on AFCS mode controller (GMC 710) prior to the following checks and reset the circuit breaker (if required) prior to the subsequent check. For each check, verify five-second flashing amber AP and YD annunciations, a three second disconnect aural alert, and FD bars remaining on display.

(a) Press the A/P DISC TRIM INTER switch on the pilot’s control yoke.

(b) Press the A/P DISC TRIM INTER switch on the co-pilot’s control yoke.

(c) In S/N's 4636633, 4636652–4636715, 4636717–4636717 only: Pull the AUTO PILOT circuit breaker. After check is complete, pull the PITCH TRIM circuit breaker and then reset both breakers.

(7) Autopilot and Yaw Damper Disconnect Checks – Abnormal Mode

Engage the autopilot by pressing the AP key on AFCS mode controller (GMC 710) prior to each of the following checks and reset the circuit breaker (if required) prior to the subsequent check. For each check, verify a continuous flashing red/white AP and a five second yellow YD annunciation, a red boxed AFCS annunciation (unless specified otherwise), and a continuous disconnect aural alert (until silenced by the A/P DISC TRIM INTER switch).

NOTE: After resetting each circuit breaker, allow time for the AHRS or GPS to initialize prior to conducting the next test.

(a) Press the STALL TEST switch on the instrument panel. Verify FD bars remain on the display [and no AFCS annunciation. Will not disconnect if Enhanced AFCS installed.].

(b) ‡Pull the PITCH TRIM circuit breaker. Verify FD bars remain on the display.‡

(c) ‡Pull the AUTOPILOT circuit breaker. Verify the FD bars remain on the display.‡

(d) Pull the GMC circuit breaker. Verify the FD bars are removed from the display [and no AFCS annunciation].

(e) Pull the INTEG AV 1 circuit breaker. Verify the FD bars remain on the display if XFR button is pointing to the right. Various CAS messages will appear due to GIA 1 powering down.

(f) Pull the INTEG AV 2 circuit breaker. Verify the FD bars remain on the display if XFR button is pointing to the left. Various CAS messages will appear due to GIA 2 powering down.

(g) Pull the AHRS 1 circuit breaker. Verify the red boxed ACFS annunciation does not appear. Verify the FD bars are removed from the display if XFR button is pointing to the left.

(h) Pull the AHRS 2 circuit breaker. Verify the red boxed ACFS annunciation does not appear. ‡Verify the FD bars are removed from the display if XFR button is pointing to the right.‡ [Verify the FD bars remain on the display if the XFR button is pointing to the left.]

(i) In S/N's PA-46-350P S/N's 4636460, 4636463–463663–4636715, 4636717–4636719; and PA-46-350T S/N's 4692134–4692140, 4692141–4692148, 4692150–4692152, 4692154–4692214:

1) Ensure the XFR button is pointing to the right. Pull the PFD 1 circuit breaker. Verify the Autopilot remains engaged. Press the XFR button to point left. [No red AFCS annunciation.

2) Ensure the XFR button is pointing to the left. Pull the MFD circuit breaker. Verify the Autopilot remains engaged. Press the XFR button to point right. [No red AFCS annunciation.]

(j) In S/N's 4636716, 4636720 and up:

1) Ensure the XFR button is pointing to the left. Pull the PFD2 circuit breaker. Verify the Autopilot remains engaged. Press the XFR button to point right. (No red AFCS annunciation or YD Disconnect)

2) Pull the AUTO PILOT circuit breaker. After check is complete, pull the PITCH TRIM circuit breaker and then reset both breakers.

(8) Yaw Damper Disconnect Checks – Manual "Normal" Method

Engage the autopilot by pressing the AP key on AFCS mode controller (GMC) prior to the following check.

(a) Press the YD button.

(b) Verify a five-second yellow YD annunciation and FD bars remaining on display.

(9) Autopilot Flap Interconnect Checks

(a) [Press FD on the GMC to clear flight directors modes.] Engage the autopilot again by pressing the AP key on the GMC). ‡Select HDG mode by pressing the HDG key on the AFCS mode controller. Select ALT mode by pressing the ALT key on the AFCS mode controller.‡

(b) Hold pitch control (yoke) where the pitch trim system does not activate.

(c) Verify the following:

Operation	Trim Wheel Action
Lower the flaps 0° to 10°	Nose Down
Lower the flaps 10° to 20°	Nose Down
Lower the flaps 20° to 36°	Nose Down
Raise the flaps 36° to 20°	Nose Up
Raise the flaps 20° to 10°	Nose Up
Raise the flaps 10° to 0°	Nose Up

(10) Autopilot Clutch Overpower Check

The GFC 700 uses electronic torque limiting as well as mechanical slip clutches to limit the maximum servo effort. When the system is on the ground, the electronic torque limiting is removed, allowing manual checks of the slip-clutch settings.

(a) In Airplanes Equipped with GSM 85A Servo Gearboxes Only:

Factory installed in S/N's 4636460, 4636463–4636612 less 4636481; and S/N's 4692134–4692202 less 4692141, 4692149 and 4692153.

1) On the ground, with the autopilot off, check for freedom of control movement in all control axes, including trim.

2) Power up the aircraft avionics.

3) Engage the Autopilot by pressing the AP key on the AFCS mode controller.

4) Manually overpower the autopilot clutches in pitch and roll. If the Autopilot clutches cannot be overpowered, check the servo clutch torque settings. See "Servo Clutch Torque Adjustment" on page 221057.

5) Actuate and hold PITCH TRIM switch in either the NOSE UP or NOSE DOWN direction to disconnect the autopilot. While the trim is running, restrain the aircraft pitch trim wheel and verify that the trim clutch can be overpowered. If it cannot be overpowered, check the trim servo clutch torque setting. See "Servo Clutch Torque Adjustment" on page 221057.

6) Engage the autopilot by pressing the AP key on the AFCS mode controller. Actuate and hold the manual electric trim switch in either the up or down direction to disconnect the autopilot. Verify that the trim wheel moves smoothly in both directions throughout the entire trim range during manual electric trim operation. If the trim wheel hesitates, this may indicate that the pitch trim clutch is slipping and proper clutch setting ("Servo Clutch Torque Adjustment" on page 221057) and cable tension should be verified. See "Troubleshooting" on page 221027; and "Pitch Trim Servo" on page 221050 and "Servo Clutch Torque Adjustment" on page 221057. If both clutch setting and cable tension are within tolerance, check the aircraft pitch trim system for excessive friction. See Rigging and Adjustment under Elevator Trim Controls in 27-30-00.

(b) In Airplanes Equipped with GSM 86 Servo Gearboxes Only:

Factory installed in S/N's 4636613 and up; and S/N's 4692203 and up. Authorized service replacement in S/N's 4636460, 4636463–4636612 less 4636481; and S/N's 4692134–4692202 less 4692141, 4692149 and 4692153.

1) On the ground, with the autopilot off, check for freedom of control movement in all control axes, including trim.

2) Power up the aircraft avionics.

3) Engage the Autopilot by pressing the AP key on the GMC.

4) Manually overpower the autopilot clutches in pitch and roll. If the Autopilot clutches cannot be overpowered, verify the correct servo clutch is installed. See "Chart 4" on page 221058 and "Figure 14" on page 221046. If the correct clutch is installed and cannot be overpowered, replace the clutch.

5) Actuate and hold PITCH TRIM switch in either the NOSE UP or NOSE DOWN direction to disconnect the autopilot. While the trim is running, restrain the aircraft pitch trim wheel and verify that the trim clutch can be overpowered. If it cannot be overpowered, verify the correct servo clutch is installed. See "Chart 4" on page 221058 and "Figure 14" on page 221046. If the correct clutch is installed and cannot be overpowered, replace the clutch.

6) [Engage the autopilot by pressing the AP key on the GMC.] Actuate and hold the manual electric trim switch in either the up or down direction to disconnect the autopilot. Verify that the trim wheel moves smoothly in both directions throughout the entire trim range during manual electric trim operation. If the trim wheel hesitates, this may indicate that the pitch trim clutch is slipping and proper clutch and cable tension should be verified. See "Troubleshooting" on page 221027; "Chart 4" on page 221058, "Figure 14" on page 221046, and "Pitch Trim Servo" on page 221050. If clutch is as specified and cable tension is within tolerance, check the aircraft pitch trim system for excessive friction. See Rigging and Adjustment under Elevator Trim Controls in 27-30-00. Replace clutch as required.

(11) Manual Electric Pitch Trim Speed Check

(a) Run the MANUAL ELECTRIC PITCH TRIM in one direction until it runs against the mechanical stop.

(b) Run the trim in the opposite direction and, using a stop watch or equivalent device, time the trim speed to the opposite mechanical stop. Verify that the elapsed time for full travel measures 45 ± 5 [40 ± 5] seconds.

(12) Autopilot Operation Checks

(a) Engage the Autopilot by pressing the AP key on the AFCS mode controller (GMC 710). Push the HDG knob to synchronize the heading bug to the current aircraft heading. Select HDG mode by pressing the HDG key on the AFCS mode controller. Verify the command bars are level and the control wheel is stationary. (There may be some roll motion in the yoke if the aircraft not perfectly level.)

(b) Turn the HDG knob to the left and right and check that the command bars move in the correct direction and the control wheel follows the command bars.

(c) Push and hold the CWS button and pull the control wheel to the center of the pitch control range [and wings level]. Release the CWS button. Verify the autopilot clutches reengage and hold the wheel stationary.

(d) Holding the control wheel lightly, move the NOSE UP/DN WHEEL on the AFCS mode controller one detent position in the UP direction to increase the pitch reference. Verify the command bars move up ~0.5 degree and the control wheel begins moving aft. (In some aircraft, the down spring may require manual assistance to get aft control stick movement).

(e) While holding the control wheel firmly, press and hold the CWS button to re-synchronize the pitch reference. Re-center the control wheel to wings level and mid-range elevator travel. Release the CWS button and check that servo clutches reengage before releasing the control wheel.

(f) Move the NOSE UP/DN WHEEL on the AFCS mode controller two detent positions in the DN direction. Verify the command bars command a pitch attitude ~1.0 degree lower and the control wheel begins moving forward. Hold the controls and press CWS to re-center the command bars and stop control wheel movement.

(g) With the Autopilot still engaged and the CWS button pressed, move the control wheel to its aft limit. Release the CWS button and apply continuous forward pressure, slowly moving the control wheel. After a brief delay, verify the trim wheel begins moving in a trim up direction.

(h) Grip the control wheel and press the CWS button. Verify trim motion stops. Move the control wheel to the forward limit and release the CWS button. Slowly move the control wheel aft. After a brief delay, verify the trim wheel begins to trim down. Relieve pressure on the wheel and verify the trim motion stops. Check that the trim wheel is free to turn. Hold the control wheel and press the A/P DISC TRIM INTER switch to disconnect the autopilot.

(13) VOR/LOC/GS Test

Check the VOR, ILS, and Glideslope functions with ramp test equipment. Operate the equipment according to the test equipment manufacturer's instructions. Adjust the RF signal to a level adequate to perform the test. Select the appropriate HSI source by using the CDI softkey.

NOTE: The PFD HSI does not show a course deviation bar unless a valid VHF NAV frequency is tuned.

(a) ‡Ensure FD is coupled to PFD1 as indicated by a left pointing arrow next to the XFR button.‡

(b) Simulate a VOR signal on a radial 360° relative to the airplane. Tune the NAV1 and NAV2 receivers to the simulation frequency. Verify FD is coupled left as indicated by a left pointing arrow on the AFCS mode controller next to the XFR button.

(c) Set the HSI on PFD1 to VOR1 by pressing the CDI soft key until VOR1 is selected. Set the HSI on PFD2 to VOR2 by pressing the CDI soft key until VOR2 is selected. Rotate CRS1 and CRS2 knobs to set VOR1 and VOR2 course pointers to 360°. (CDI Sync must be "OFF" on MFD Aux Group, page 5.)

(d) Verify full scale deflection of VOR1 and VOR2 CDI by varying CRS1 and CRS2 selected course at least 10° left and right. Reset course pointers to 360°.

(e) Engage the autopilot and press the NAV key on the AFCS mode controller. Using the CRS1 knob, alter course by 10° to the right. Verify the Flight Director and aircraft controls respond by flying to the VOR course. Repeat to the left.

(f) Couple FD to PFD2 by pressing the XFR button on the AFCS mode controller. Verify FD is coupled right as indicated by a right pointing arrow on the AFCS mode controller next to the XFR button. Verify that the NAV mode extinguishes, then returns to default PIT and ROL mode.

(g) Repeat step (e) using CRS2 knob while coupled to PFD2. [Then disconnect the autopilot.]

(h) Set CRS1 and CRS2 course pointers to 360° relative to the airplane.

(i) Simulate a Localizer/Glideslope signal. Tune this signal on NAV1 and NAV2 receiver. Set the PFD1 HSI to LOC1 and PFD2 HSI to LOC2 by pressing CDI soft key until LOC1 and LOC2 is selected. Use the test equipment to center the deviation bars (localizer and glideslope) on PFD1 and PFD2.

(j) [Engage the autopilot by pressing AP button on the GMC.] Press the APR key on the AFCS mode controller. Verify that the LOC and GS annunciations are green on PFD1 and PFD2. Apply right/left and up/down localizer/glideslope signals using the test equipment. Verify that the Flight Director and flight controls respond appropriately, while coupled to PFD2.

(k) Couple FD to PFD1 by pressing the XFR button on the AFCS mode controller. Verify FD is coupled to PFD1 as indicated by a left pointing arrow on the AFCS mode controller next to the XFR button. [Press APR button and verify the LOC and GS annunciations are green on PFD1 and PFD2.]

(l) Apply right/left and up/down localizer/glideslope signals using the test equipment. Verify that the Flight Director and flight controls respond appropriately, while coupled to PFD1.

(14) ESP Functional Check

This procedure is required only for airplanes equipped with the optional Enhanced AFCS features.

NOTE: This check requires valid GPS reception.

(a) On the MFD, navigate to the SYSTEM SETUP page of the AUX group using the FMS knob. Press SETUP 2 softkey.

(b) Verify that there is a window for Stability & Protection and the status is “ENABLED”.

(c) Press MSG softkey. Verify that there are no “ESP FAIL” or “ESP OFF” system messages on the PFDs.

(d) Cover both GPS1 and GPS2 antennas.

(e) Verify on PFD 1 and PFD 2 the ESP Roll Indices are displayed at 45° on the roll indicator.

(f) On the MFD, activate the cursor then rotate the large FMS knob to select Stability & Protection.

(g) Rotate the small FMS knob to change the status to “DISABLED”.

(h) Verify on PFD 1 and PFD 2 the ESP Roll Indices are not displayed at 45° on the roll indicator on the Attitude Display.

(i) Press MSG {Message} softkey on PFD1 & PFD2. Verify that there is an “ESP OFF” system message on both PFDs.

(j) Remove the cover from both GPS1 and GPS2 antennas.

(k) Remove power to the aircraft.

(l) Wait approxilmately 15 seconds before re-applying aircraft power.

(m) Verify MFD splash screen states “Enhanced Stability Protection”.

(n) Once AFCS PFT is complete and prior to GPS acquisition, verify on PFD1 and PFD2 the ESP Roll Indices are displayed at 45° on the roll indicator.

(o) After GPS has acquired satellites, verify on PFD1 and PFD2 the ESP Roll Indices are not displayed at 45° on the roll indicator on the Attitude Display.

(p) On the MFD, navigate to the SYSTEM SETUP page of the AUX group using the FMS knob.

(q) Press Setup 2 softkey. Verify there is a window for Stability & Protection and the status is “ENABLED”.

(r) Press MSG softkey on PFD 1 and PFD 2. Verify that there are no “ESP FAIL” or “ESP OFF” system messages on the PFDs.

D. Inspections

(1) 100 Hours

In airplanes equipped with GSM 85A servo gearboxes only, each 100 hours, remove the pitch, pitch trim, roll, and yaw servos, and inspect slip clutch torque per “Servo Clutch Torque Adjustment” on page 221057.

(2) Twelve (12) Months

(a) Perform a visual inspection of the GFC 700 system.

1) Inspect the servos, connectors, support structures, and bridle cables to ensure that no corrosion, chafing, cracks, or other defects exist.

2) Check the servo bridle cables to ensure no fraying, corrosion, or other damage exists. See Control Cable Inspection in 27-00-00. Replace the cable if the condition is questionable. Check the tension of each servo bridle cable per “Figure 15” on page 221049, “Figure 16” on page 221051, “Figure 17” on page 221053, or “Figure 18” on page 221055, as appropriate.

3) Inspect the GFC 700 system wiring to ensure no chafing, wear, or other damage exists.

(b) Manually move the ailerons (for roll servo), elevators (for pitch servo), elevator trim wheel (for pitch trim servo), and rudder pedals (for yaw servo) from stop to stop and observe the servo, capstan, and control surface rigging. Ensure there is no binding in the bridle cabling, and that the capstan pulleys rotate freely.

(c) Manually override each servo gearbox slip clutch every 12 months. See “Autopilot Clutch Overpower Check” on page 221032.

(3) First 1,000 Hours or Three (3) Years

At the first 1,000 hours or three (3) years; or any time the GSA 8X servo actuator is separated from the servo gearbox: grease the servo actuator output gear with AeroShell 33MS.

E. Components

(1) GMC 710 AFCS Control Unit

See “Figure 10”.

GMC 710 AFCS Control Unit
Figure 10

(a) Description

The dedicated AFCS controls on the GMC 710 allow crew control interface with the various GFC 700 autopilot / flight director functions. The GMC 710 is powered by the #1 avionics bus.

(b) Troubleshooting

See “Chart 2” on page 221038.

(c) Removal

1) Use a 3/32” hex drive tool to turn each of the four locking sockets ¼ turn counterclockwise until they reach their stops.

2) Disconnect backshell assembly from unit.

(d) Installation

1) Inspect connector(s) for damaged pins.

2) Connect backshell assembly to unit.

3) Hold unit flush with the instrument panel, ensuring locking stud alignment marks are in the vertical position.

4) Use a 3/32” hex drive tool to turn each of the four locking sockets ¼ turn clockwise. This may require applying a small amount of forward pressure to engage the ¼ turn sockets.

(e) Return to Service

1) Original GMC 710 Reinstalled

No software or configuration loading is required if the removed GMC 710 is re-installed. Continue to the GMC 710 Test under LRU Test Procedures in 34-25-01 (G1000) or 34-25-02 (G1000 NXi), as appropriate.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New Repaired or Exchange GMC 710 Installed

If a new repaired or exchange GMC 710 is installed, the correct software and configuration files must be loaded to the unit. See G1000/G1000 NXi Software/Configuration Procedure in 34-25-01 (G1000) or 34-25-02 (G1000 NXi) and then continue to the GMC 710 Test under LRU Test Procedures in 34-25-01 (G1000) or 34-25-02 (G1000 NXi), as appropriate.

CHART 2
TROUBLESHOOTING / MESSAGE ADVISORIES - GMC 710

Failure Message	Cause	Solution
GMC CNFG – GMC Config error. Config service req'd.	The G1000/G1000 NXi has detected a GMC 710 configuration mismatch.	Load GMC configuration files. Replace GMC. If problem persists, replace master configuration module(1), check config module wiring for faults and replace if necessary.
GMC FAIL – GMC is inoperative.	The G1000/G1000 NXI has detected a failure in the GMC 710.	Check wiring for faults per "Failed Path Messages" under Troubleshooting in 34-25-01 or 34-25-02, as appropriate. Replace the GMC 710.
MANIFEST – GMC software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GMC 710.	Load the correct software version. See G1000 (NXi) Software/Configuration Procedure, under Software Files in 34-25-01 or 34-25-02, as appropriate.
GMC KEYSTK – GMC [key name] key is stuck.	A key is stuck on the GMC bezel.	Exercise the key to free it. If the problem persists, replace the GMC 710.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000/G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

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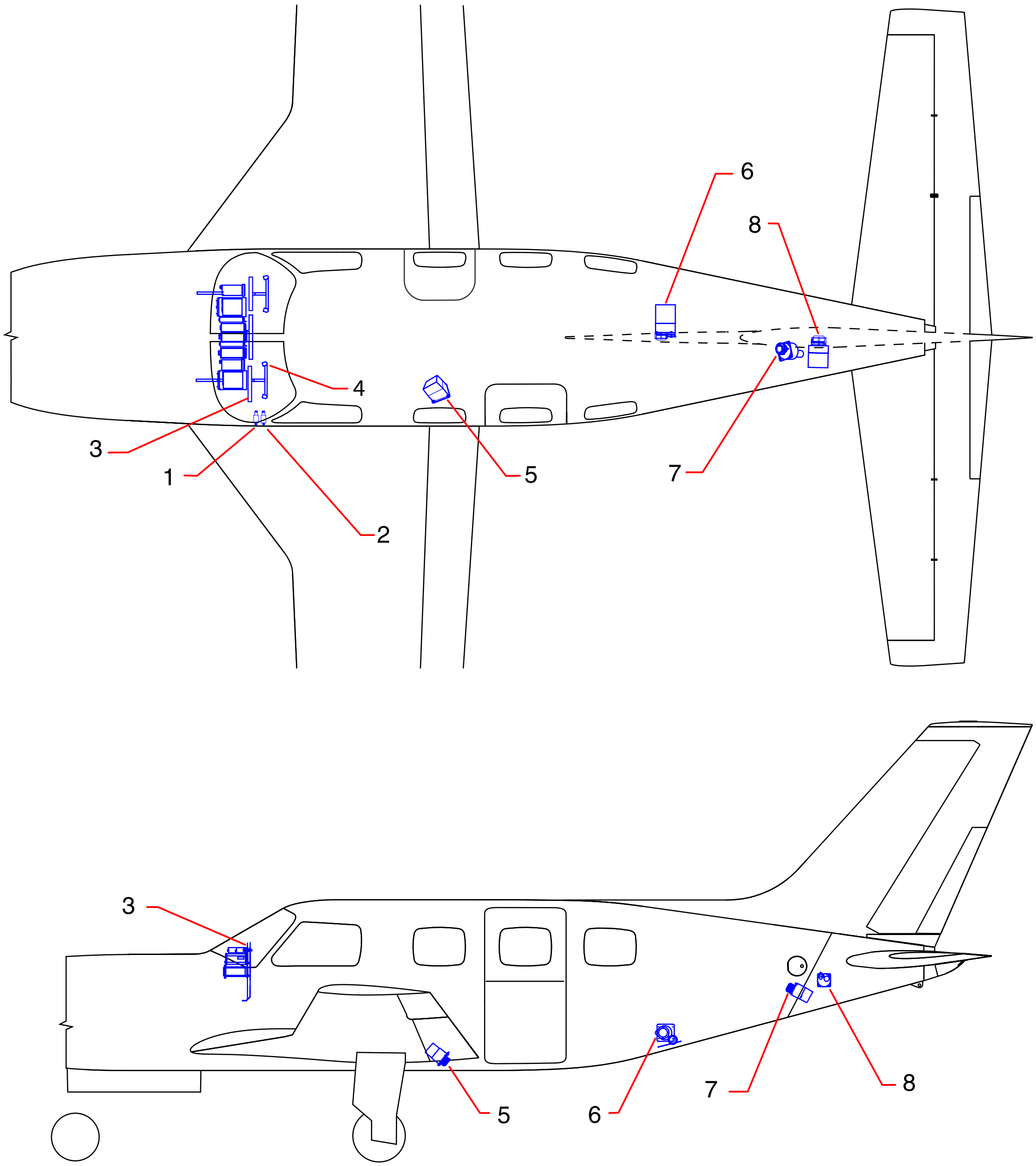


1. CB333 CIRCUIT BREAKER (5 AMP) (REF)
2. CB334 CIRCUIT BREAKER (5 AMP) (REF)
3. EFIS DISPLAYs (PFD’s, MFD) (REF)
4. CONTROL WHEEL SWITCH INSTALL (BOTH SIDES)
5. ROLL SERVO
6. TRIM SERVO
7. YAW SERVO
8. PITCH SERVO

Autopilot Component Locator
Figure 11

(2) Garmin Servos

The Garmin servos are a two part system the Servo Actuator (i.e., motor) and Servo Gearbox. See “Figure 12” on page 221042.

(a) GSA 8X Servo Actuator

The GSA 8X servo actuator is an electromechanical unit that provides automatic control of a single flight axis (pitch, roll, or yaw), or their associated trims. The GSA 8X receives serial RS-485 data packets from two GIA XX Integrated Avionics units, which contain data from the Flight Director, the GRS XX AHRS, and the GDC XX Air Data Computer. The drive clutch solenoid engages the GSM 85A drive clutch. The GSA 8X is mounted to the Garmin Servo Gearbox and can be removed and replaced without removing or de-rigging the servo gearbox.

1) Removal

a) Turn the master switch off. Pull the AUTOPILOT (and, if appropriate, PITCH TRIM) circuit breaker(s).

b) Disconnect the servo actuator electrical harness connector.

c) Remove the bolts and washers securing the servo actuator to the servo gearbox and remove the servo actuator.

2) Installation

a) Grease the servo actuator output gear with AeroShell 33MS.

NOTE: Two O-ring seals are pressed into grooves in the back plate of the servo gearbox units. If during installation, removal, or reinstallation of a servo gearbox either of these O-rings becomes detached from the rear plate, carefully inspect and ensure there are no cuts or other damage to the O-ring(s). If no damage is found, reinstall the O-ring(s) in the rear plate, being careful not to stretch or compress it along its length while installing into the groove in the rear plate. If either O-ring is missing or damaged, replace with Garmin P/N 251-20031-00.

b) Position the servo actuator to the servo gearbox, secure with bolts and washers (4 ea.) as follows:

1] Insert all bolts and thread in approximately two turns by hand.

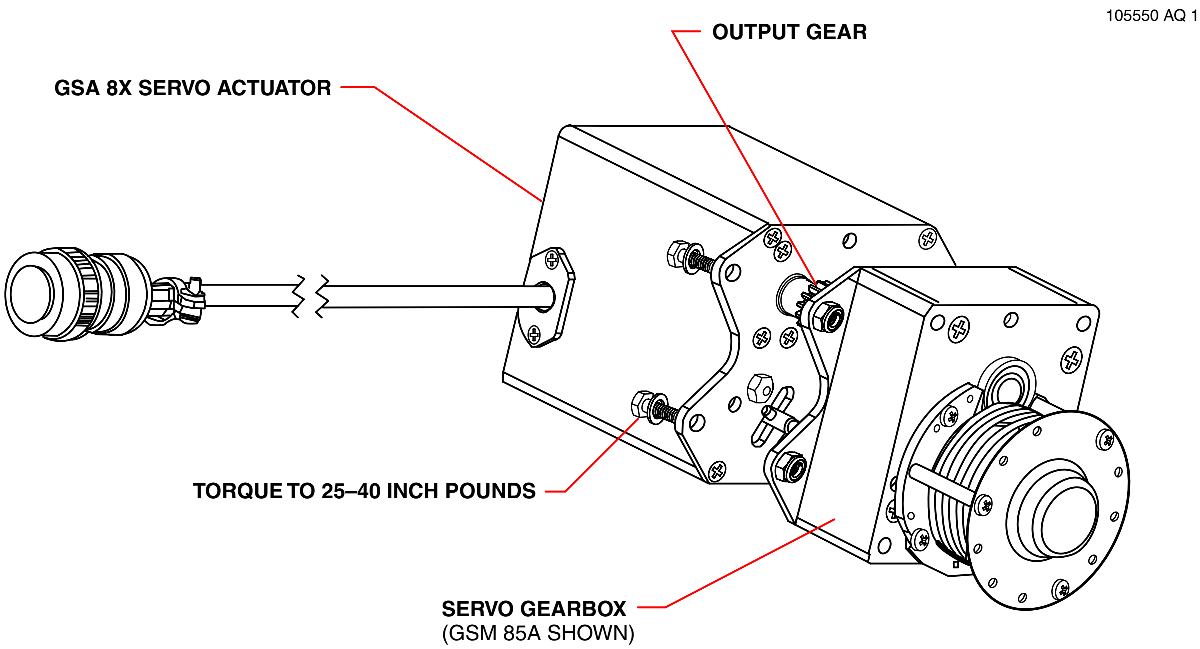
2] Starting with two opposite corner bolts, tighten all bolts until 1/64” to 1/32” (½ to 1 turn) from fully seated.

3] Verify that the GSA 8x is still slightly loose relative to the servo gearbox, ensuring there is still relative movement between the two units.

4] Starting with two opposite corner bolts, tighten all bolts to proper torque per “Figure 12” on page 221042.

c) Connect the servo actuator electrical harness connector.

d) Push-in the AUTOPILOT (and, if appropriate, PITCH TRIM) circuit breaker(s).



GSA 8X Actuator / GSM 8X Gearbox Assembly
Figure 12

(b) GSM 85A Servo Gearbox

Factory installed in S/N's 4636460, 4636463–4636612 less 4636481; and S/N's 4692134–4692202 less 4692141, 4692149 and 4692153.

The GSM 85A servo mount is mounted to the aircraft structure, via a custom mounting bracket, and is responsible for transferring the output torque of the GSA 8X servo actuator to the mechanical flight control surface linkage. The GSM 85A houses its internal components within an extruded aluminum housing, like the GSA 8X. There are multiple variations of the GSM 85A, all designed to accommodate various flight-control system interfaces and torque requirements. The slip-clutch settings and type of output capstan are described later in this chapter.

Removal and Installation procedures are under the individual functional servo paragraphs, below.

(c) GSM 86 Servo Gearbox

Factory installed in S/N's 4636613 and up; and S/N's 4692203 and up. Authorized service replacement in S/N's 4636460, 4636463–4636612 less 4636481; and S/N's 4692134–4692202 less 4692141, 4692149 and 4692153; see Note before installing as service replacement.

NOTE: The Garmin G1000 system software delivered from the factory in S/N's 4636460, 4636463–4636509; and S/N's 4692134–4692171 is not compatible with GSM 86 servo gearboxes. Ensure G1000 software version 720.14 or later has been installed per Piper Service Letter 1148 or 1179 prior to installing GSM 86 servo gearboxes.

The GSM 86 servo gearbox is mounted to the aircraft structure, via a custom mounting bracket, and is responsible for transferring the output torque of the GSA 8X servo actuator to the mechanical flight control surface linkage. The GSM 86 houses its internal components within an extruded aluminum housing like the GSA 8X. There are multiple variations of the GSM 86, all designed to accommodate various flight-control system interfaces and torque requirements.

The GSM 86 uses a preset (non-adjustable) slip clutch rather than the adjustable friction disc clutch used in the GSM 85A. This provides a tighter tolerance slip torque setting that supports advanced GFC 700 features and a significantly increased interval between required inspections which reduces maintenance costs. It is comprised of three main parts: gearbox, clutch kit (available in a variety of torque settings), and capstan kit.

Removal and Installation procedures are under the individual functional servo paragraphs, below.

(d) Return to Service

1) Original Servo(s) Reinstalled

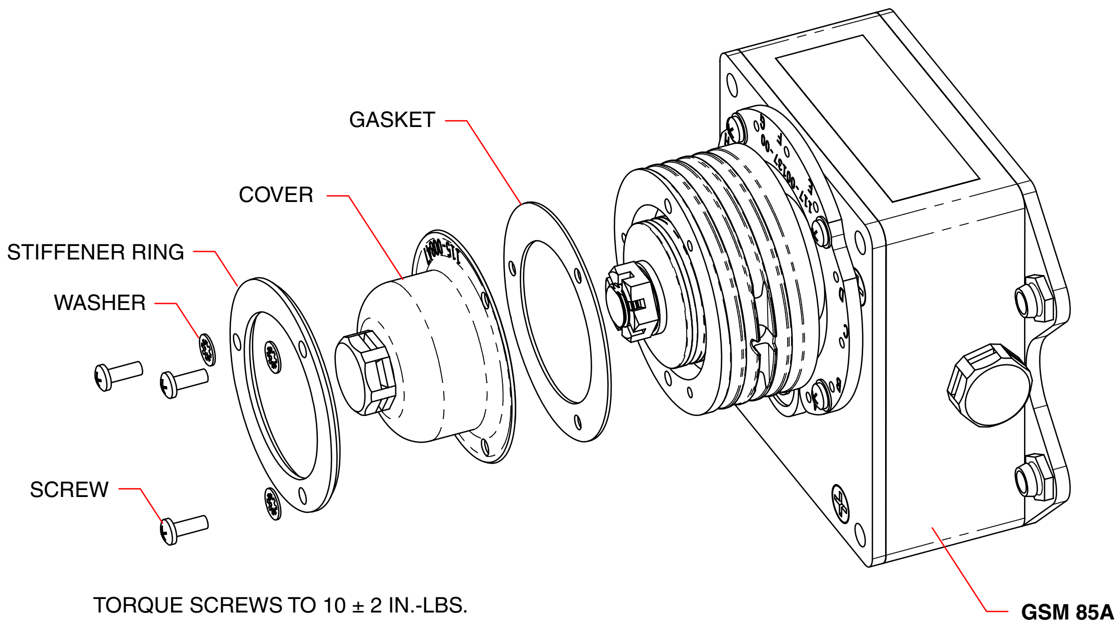
No software loading is required if the removed servo(s) is re-installed. Continue to “Ground Checks” on page 221027.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

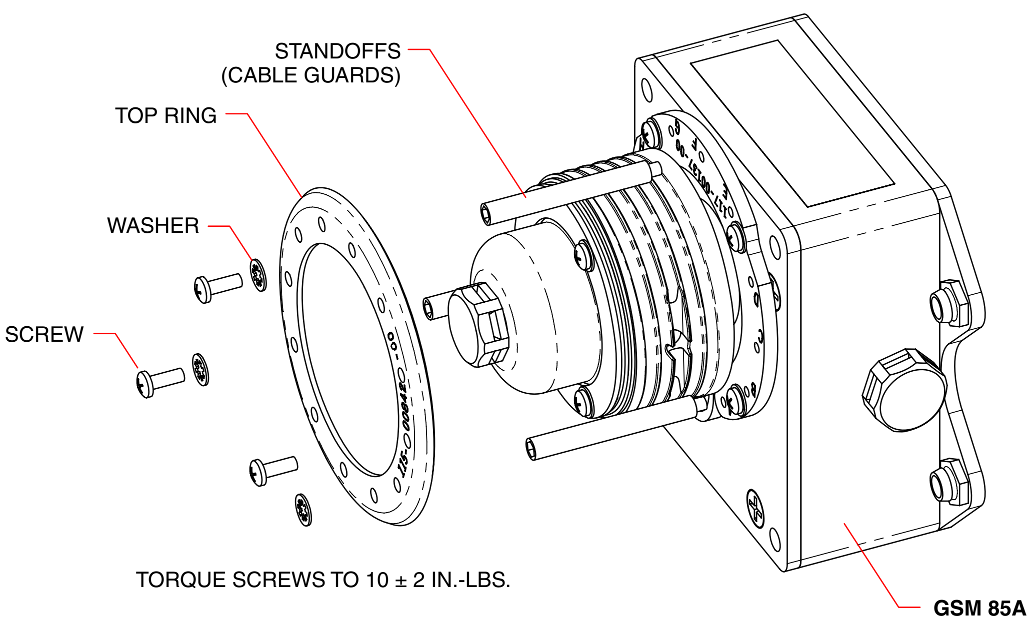
2) New Repaired or Exchange Servo(s) Installed

If a new repaired or exchange servo is installed, the correct software files and certification gains must be loaded to the unit. See Software/Configuration Procedure in 34-25-01 (G1000) or 34-25-02 (G1000 NXi) and then continue to the Return-to-Service Procedure in 34-25-01 (G1000) or 34-25-02 (G1000 NXi).

3) If no other service is to be performed, continue to “Ground Checks” on page 221027.

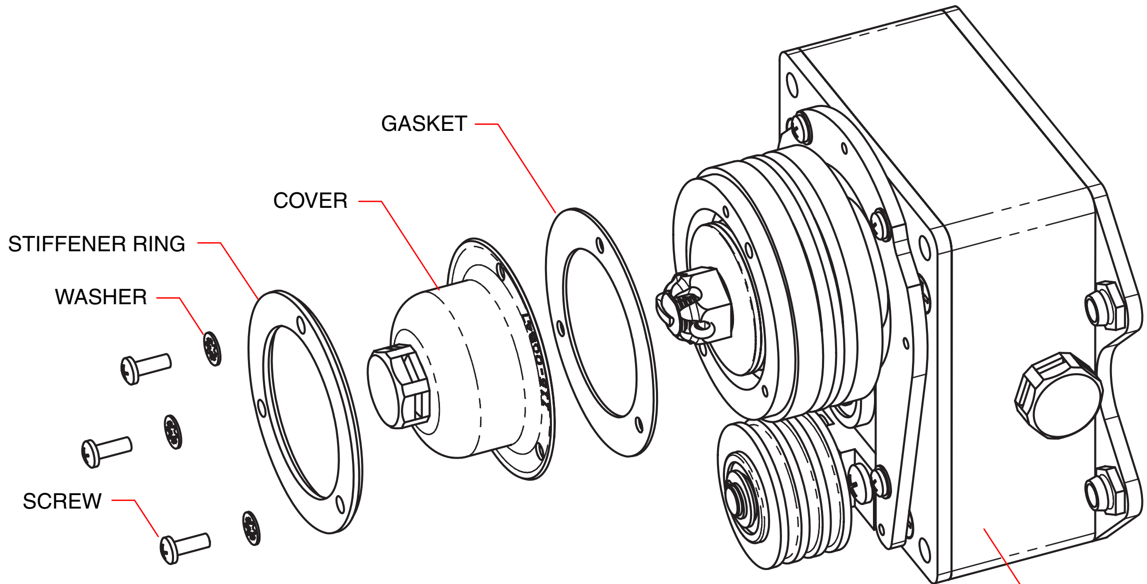


NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.



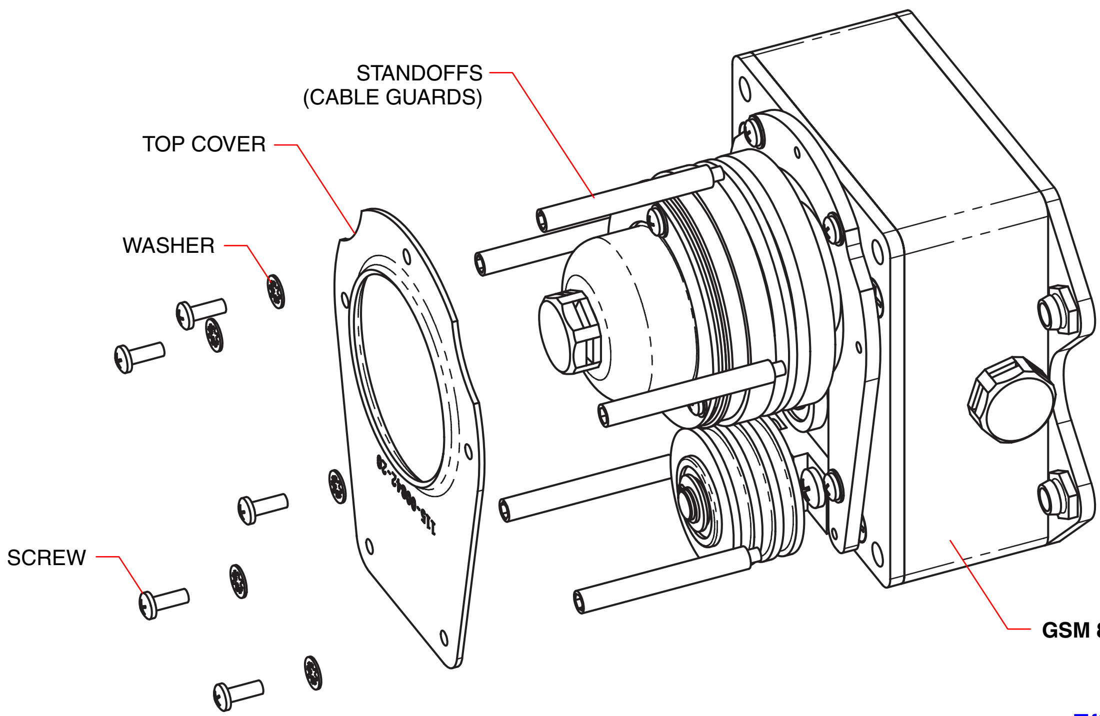
Effectivity
4636460, 4636463–4636612
less 4636481; and
4692134–4692202 less 4692141,
4692149 and 4692153

GSM 85A Servo Gearbox
Figure 13 (Sheet 1 of 2)



TORQUE SCREWS TO 10 ± 2 IN.-LBS.

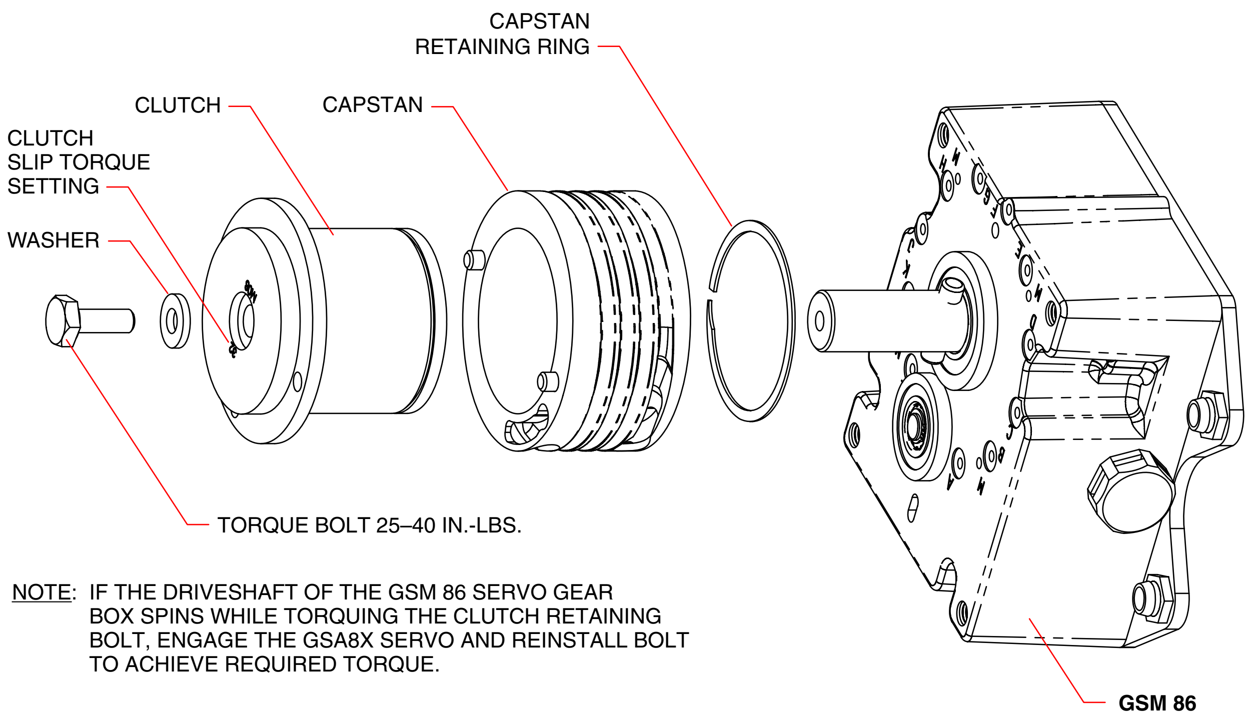
NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.



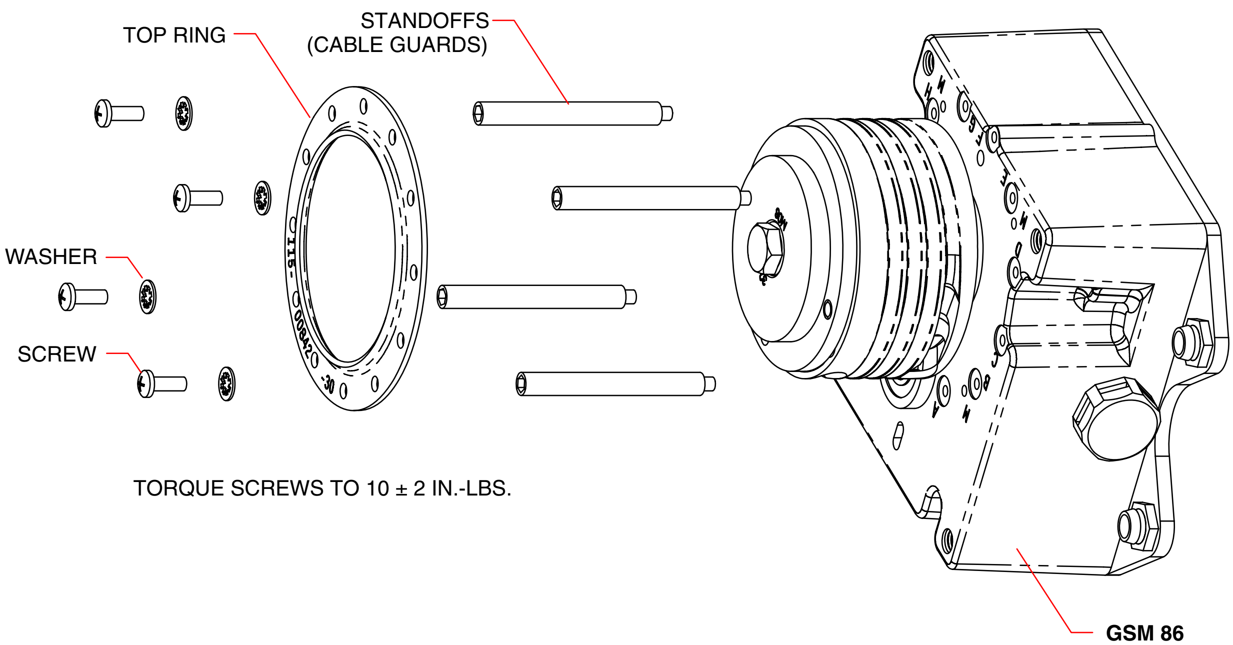
TORQUE SCREWS TO 10 ± 2 IN.-LBS.

Effectivity
4636460, 4636463–4636612
less 4636481; and
4692134–4692202 less 4692141,
4692149 and 4692153

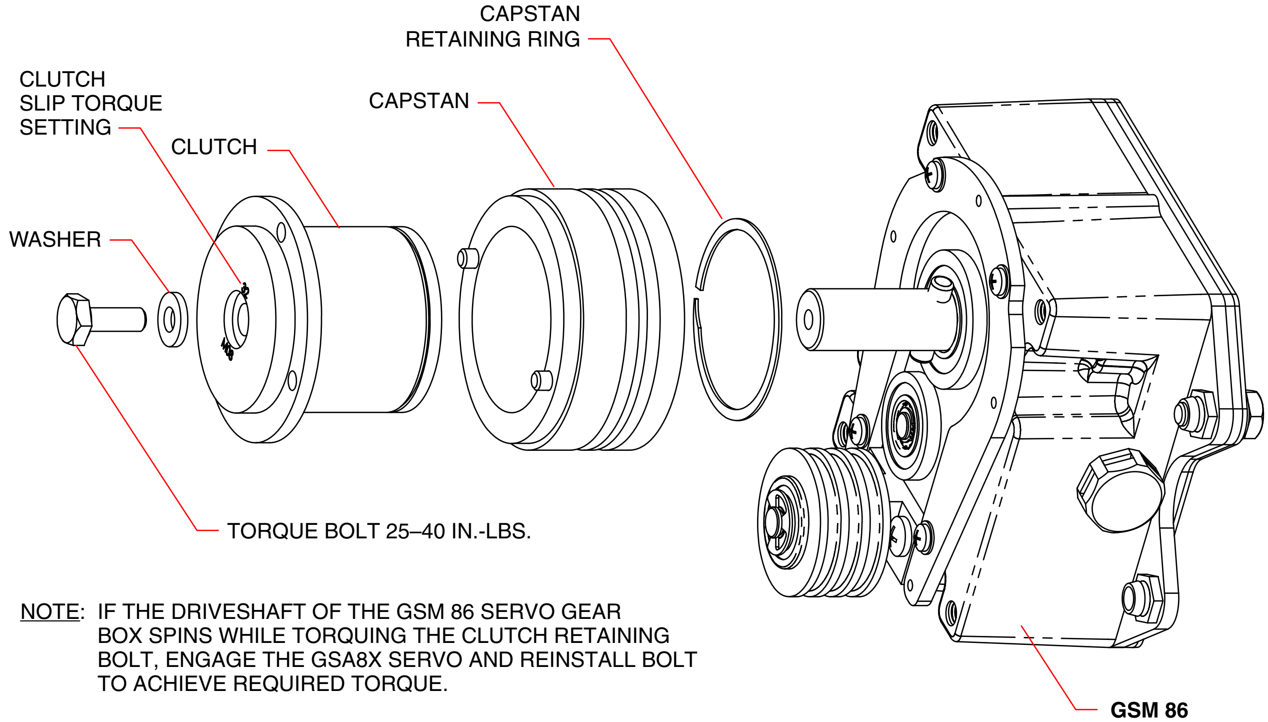
GSM 85A Servo Gearbox with Idler
Figure 13 (Sheet 2 of 2)



NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.

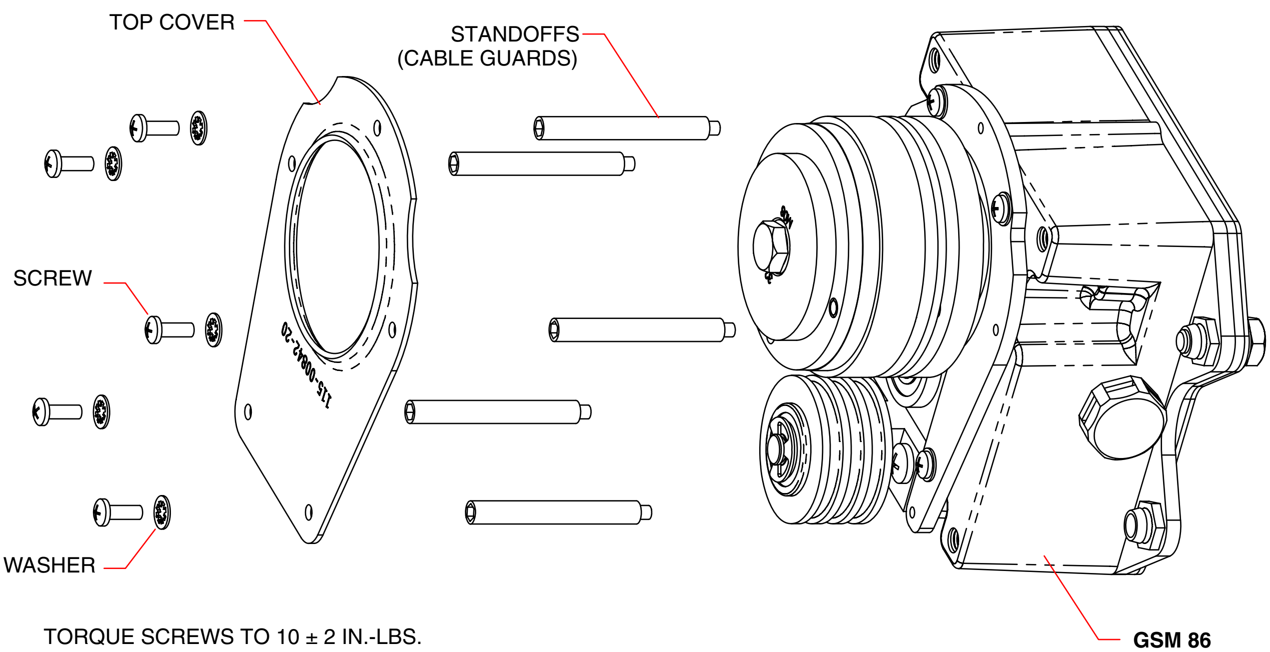


GSM 86 Servo Gearbox
Figure 14 (Sheet 1 of 2)



NOTE: IF THE DRIVESHAFT OF THE GSM 86 SERVO GEAR BOX SPINS WHILE TORQUING THE CLUTCH RETAINING BOLT, ENGAGE THE GSA8X SERVO AND REINSTALL BOLT TO ACHIEVE REQUIRED TORQUE.

NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.



TORQUE SCREWS TO 10 ± 2 IN.-LBS.

GSM 86 Servo Gearbox with Idler
Figure 14 (Sheet 2 of 2)

(3) Roll Servo

See “Figure 11” on page 221040, “Figure 12” on page 221042, and “Figure 15” on page 221049.

The roll servo is mounted underneath the left aft facing passenger seat or the entertainment cabinet, whichever is installed.

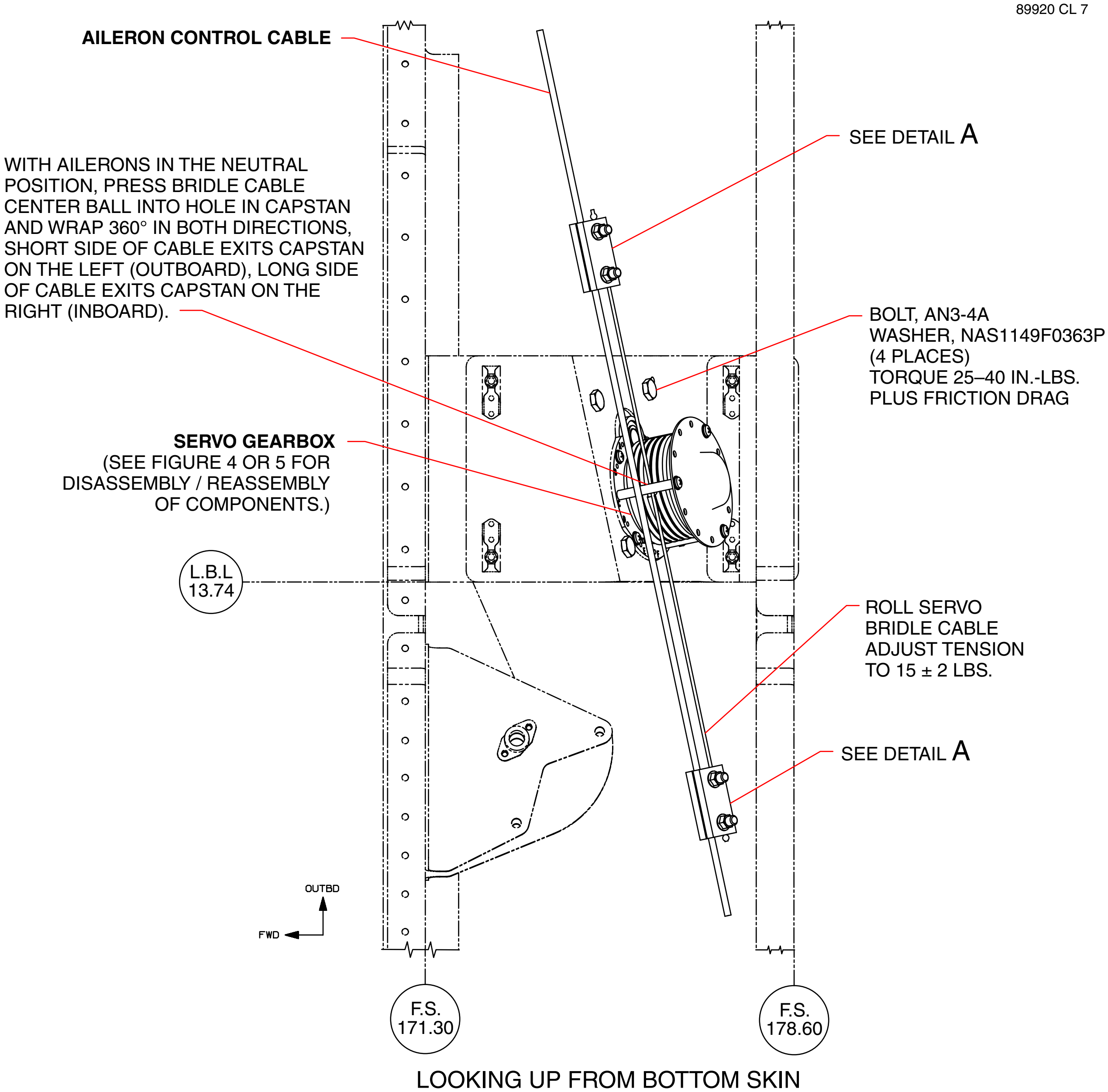
(a) Removal

1) Remove the left aft facing passenger seat.
2) Remove adjacent carpet.
3) Disconnect autopilot harness connector.
4) Remove nuts and screws (2 ea.) securing each cable clamp (2) and remove cable clamps from aileron balance and autopilot bridle cables.
5) Remove bolts and washers (4 ea.) securing roll servo (GSA 8X/GSM 8X) to mounting brackets and remove roll servo with attached bridle cable.

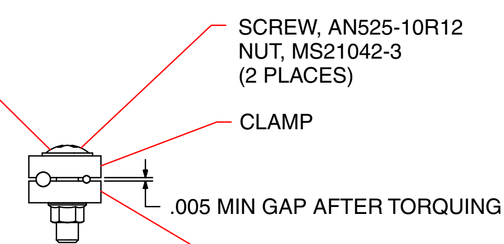
(b) Installation

NOTE: If installing a new, repaired, or exchanged GSM 85A servo gearbox, adjust slip-clutch torque per “Servo Clutch Torque Adjustment” on page 221057 before installation.

1) Rig ailerons per Aileron Control Rigging and Adjustment, 27-10-00.
2) Lock the control wheels together in the aileron neutral position using a suitable tool.
3) Lock the ailerons in neutral position using a suitable contour fixture at the inboard ends of the ailerons. Neutral position is defined by wing loft contour at this location.
4) Wrap autopilot bridle cable, as described in “Figure 15” and align capstan.
5) Replace cable guards and capstan cover, secure with screws. Torque screws per “Figure 13” on page 221044 or “Figure 14” on page 221046.
6) Position servo (GSA 8X/GSM 8X) as shown in “Figure 15” and install and secure servo to mounting brackets with washers and bolts (4 ea.). Torque bolts per “Figure 15”.
7) Position cable clamps (2) as shown in “Figure 15” and tighten nuts and screws (2 ea.). Adjust cable clamps in or out along the aileron balance cable to obtain the bridle cable tension specified in “Figure 15”. Torque cable clamp screws per “Figure 15”, Detail A..
8) Remove the locking fixtures. Aileron neutral position should be maintained at 0 ± 1 degree with the control wheels in neutral.
9) Use the pilot’s control wheel to move the ailerons and verify aileron travel at the primary stops (at the aileron drive sectors on the wing spars) remains as specified in 27-00-00, Chart 2.
10) Use the pilot’s control wheel to hold the ailerons firmly against the primary stops. Verify the secondary aileron stops (on the forward pressure bulkhead) retain a cushion (gap) of 0.09 ± 0.03 inch in each direction.
11) If either step “8)” thru “10)” reveal the aileron controls out of rig, repeat steps “1)” thru “10)”, above.
12) Connect autopilot harness.
13) Check aileron controls for free and correct movement.
14) Review “Return to Service” on page 221043 criteria under Garmin Servos, to determine if software needs to be reloaded.
15) Check autopilot system for binding and proper operation, see “Ground Checks” on page 221027.
16) Replace carpeting.
17) Replace the left aft facing passenger seat.



PRIOR TO ADJUSTING PRIMARY CONTROL CABLE TENSION, LOOSEN HARDWARE OF THE AUTOPILOT SERVO BRIDLE CABLE CLAMPS ENOUGH TO ALLOW THE CLAMPS TO SLIDE ON THE PRIMARY CONTROL CABLE. AFTER THE PRIMARY CONTROL CABLE TENSION IS ADJUSTED TO SPECIFICATION, READJUST AUTOPILOT SERVO BRIDLE CABLE TENSION AS SPECIFIED AND RE-TORQUE CABLE CLAMP HARDWARE.



DETAIL A

Roll Servo Installation
Figure 15

(4) Pitch Trim Servo

See “Figure 11” on page 221040, “Figure 12” on page 221042, and “Figure 16” on page 221051.

The pitch trim servo is located on the centerline underneath the baggage compartment floor.

(a) Removal

1) Remove rear seats. Remove carpet and access panels from baggage compartment floor.
2) Remove the centerline access panels in the cabin floor to expose the plastic tube that protects the trim cables.
3) Remove the pins from both ends of the plastic tube.
4) Slide the tube forward.
5) Rotate the elevator trim control wheel until the right trim cable turnbuckle clears the aft end of the tube.
6) Tie a pull rope to the right trim cable forward of the exposed turnbuckle.
7) Tie a pull rope to the right trim cable aft of the pulley.
8) Tie-off both pull ropes to maintain tension on the left trim cable and the extreme forward and aft segments of the right trim cable (i.e., forward of turnbuckle and aft of pulley).

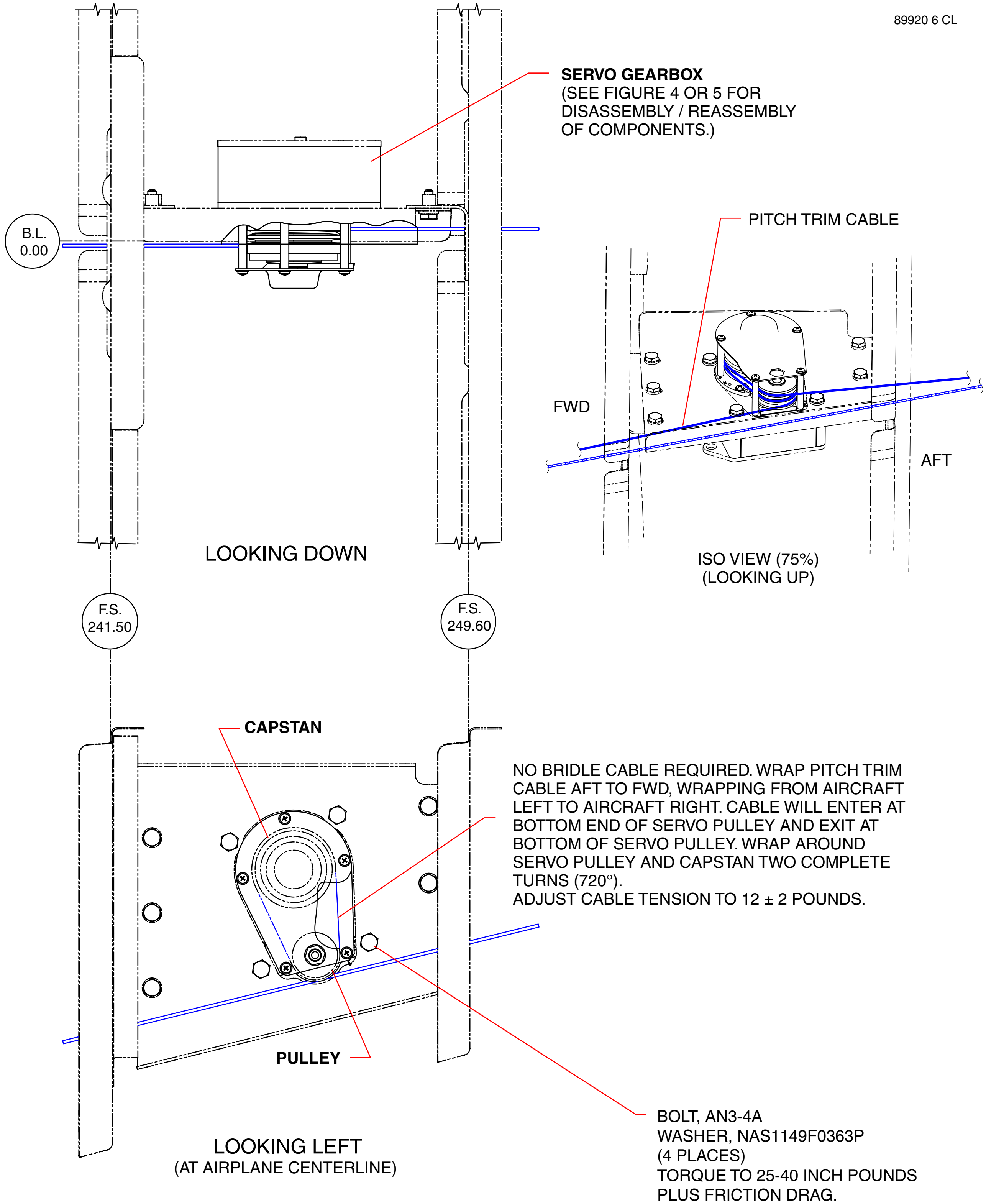
NOTE: Tension must be applied to the trim cables to prevent the cables from unwrapping from the trim wheel drum or the trim actuator, and to prevent the cables from fouling at any of the pulleys.

9) Slack off the exposed turnbuckle to relieve tension on the right trim cable segment between the turnbuckle and pulley.
10) Disconnect the autopilot harness.
11) Remove the capstan cover and cable guards (5) by removing the retaining screws (5).
12) Remove the cable from capstan and pulley.
13) Remove the bolts and washers (4 ea.) securing the trim servo (GSA 8X/GSM 8X) to its mounting bracket and remove the trim servo.

(b) Installation

NOTE: If installing a new, repaired, or exchanged GSM 85A servo gearbox, adjust slip-clutch torque per “Servo Clutch Torque Adjustment” on page 221057 before installation.

1) With capstan cover and cable guards removed, position trim servo (GSA 8X/GSM 8X) as shown in “Figure 16” on page 221051. Secure with (4 ea.) bolts and washers. Torque bolts per “Figure 16”.
2) Drape the slack left trim cable over the servo capstan.
3) Wrap the cable around the capstan and pulley two (2) complete turns while pressing the trim cable into its groove.
4) Tighten turnbuckles and remove tie ropes.
5) Install capstan cover and cable guards. Torque screws per “Figure 13” on page 221044 or “Figure 14” on page 221046.
6) Rig elevator trim per Elevator Trim Controls Rigging in 27-30-00.
7) Connect the autopilot harness.
8) Review “Return to Service” on page 221043 criteria under Garmin Servos, to determine if software needs to be reloaded.
9) Check elevator trim controls for free and correct movement.
10) Check autopilot system for proper operation, see “Ground Checks” on page 221027.



Pitch Trim Servo Installation
Figure 16

11) Replace access panels and carpet in cabin and baggage compartment floors.

12) Replace rear seats.

(5) Yaw Servo

See “Figure 11” on page 221040, “Figure 12” on page 221042, and “Figure 17” on page 221053.

The yaw servo is located in the aft fuselage/tailcone.

(a) Removal

NOTE: To remove the yaw servo from the airplane the pitch servo will.need to be removed first. See “Removal” on page 221054 under Pitch Servo.

1) Remove the tailcone access panel (see #1, Figure 3, 6-00-00).

2) Disconnect autopilot harness.

3) Remove nuts, washers, and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from rudder cables and autopilot bridle cables.

4) Remove bolts and washers (4 ea.) securing yaw servo (GSA 8X/GSM 8X) to mounting bracket and remove yaw servo with attached bridle cable.

(b) Installation

NOTE: If installing a new, repaired, or exchanged GSM 85A servo gearbox, adjust slip-clutch torque per “Servo Clutch Torque Adjustment” on page 221057 before installation.

1) Rig rudder controls per Rudder Control Rigging, 27-20-00.

2) Lock rudder in its neutral (streamlined) position.

3) Remove screws (3) and remove capstan cover and cable guards from servo.

4) Wrap autopilot bridle cable, as shown in “Figure 17”.

5) Replace cable guards and capstan cover, secure with screws (3). Torque screws per “Figure 13” on page 221044 or “Figure 14” on page 221046.

6) Position yaw servo (GSA 8X/GSM 8X) as shown in “Figure 17” and secure with bolts and washers (4 ea.). Torque bolts per “Figure 17”.

7) Attach left and right autopilot bridle cables to the left and right rudder cables with cable clamps (2). Center ball of the cable must be positioned as shown in “Figure 17”. Torque cable clamp bolts as specified in “Figure 15” on page 221049, Detail A.

8) Connect autopilot harness.

9) Remove rudder lock.

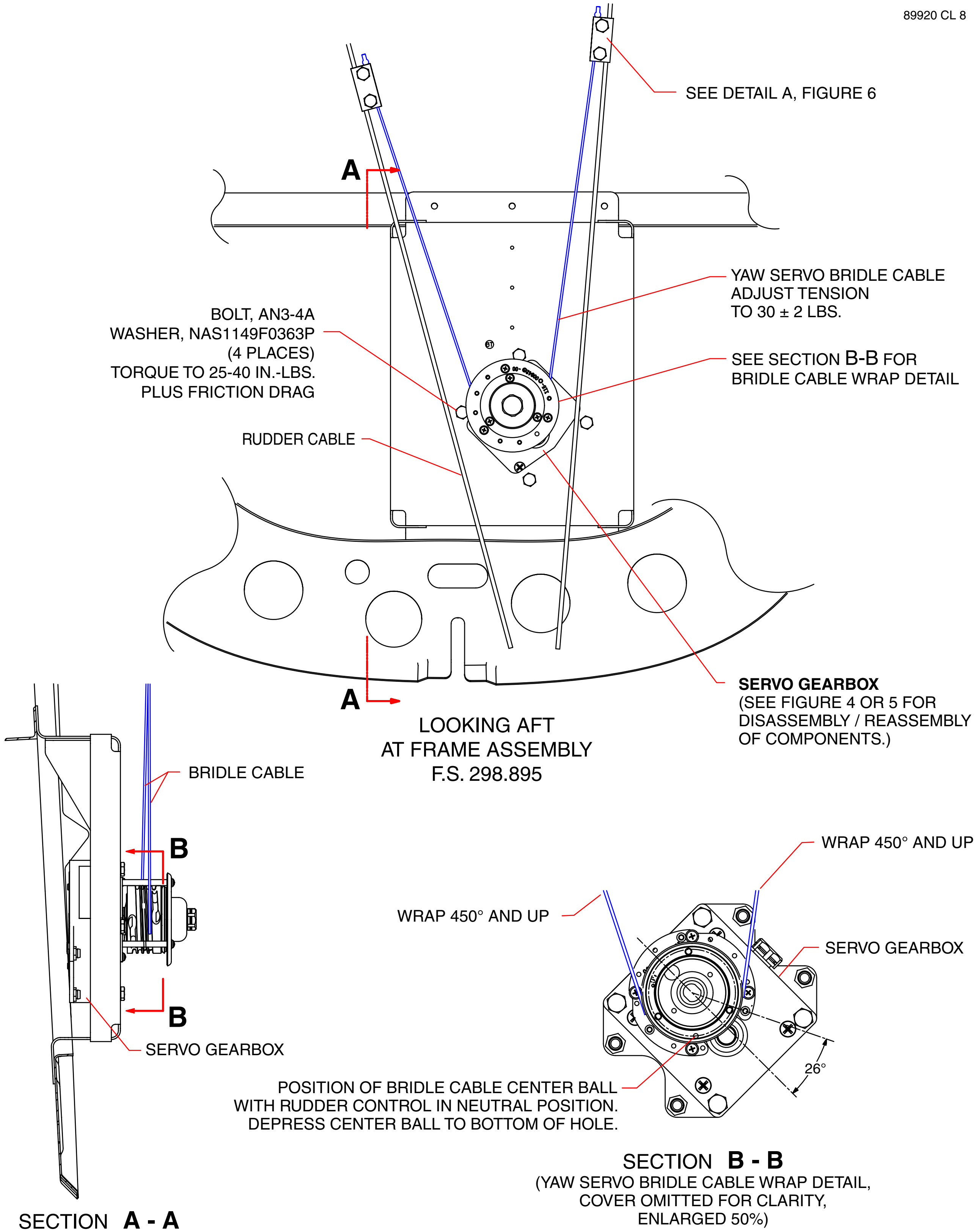
10) Check rudder controls for free and correct movement.

11) Review “Return to Service” on page 221043 criteria under Garmin Servos, to determine if software needs to be reloaded.

12) Reinstall pitch servo per “Installation” on page 221054 under Pitch Servo.

13) Check autopilot system for binding and correct operation. See “Ground Checks” on page 221027.

14) Replace the tailcone access panel and secure with twenty-three (23) screws and washers.



Yaw Servo Installation
Figure 17

(6) Pitch Servo

See “Figure 11” on page 221040, “Figure 12” on page 221042, “Figure 18” on page 221055, and “Figure 19” on page 221056.

The pitch servo is located in the aft fuselage/tailcone.

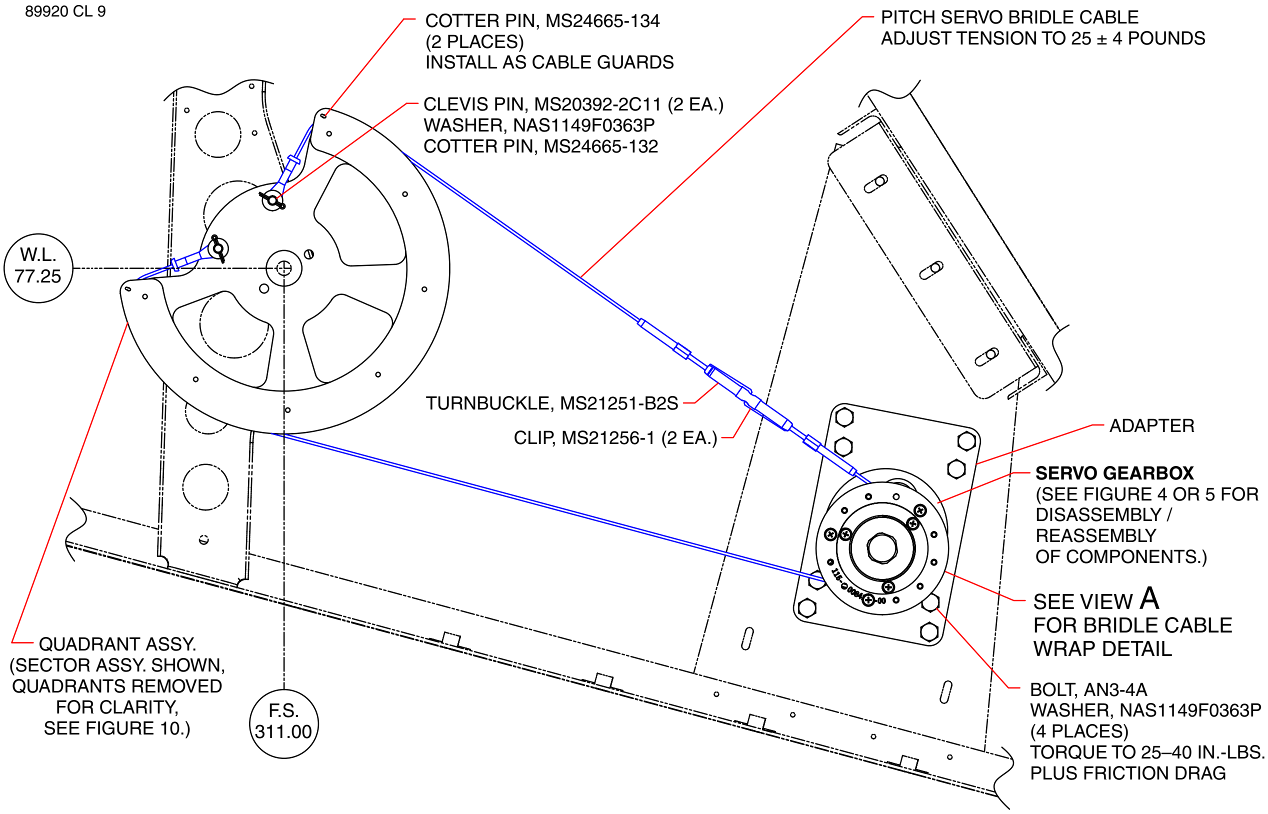
(a) Removal

1) Remove the tailcone access panel (See #1, Figure 3, 6-00-00).
2) Disconnect autopilot harness.
3) Remove locking clips (2) from turnbuckle in autopilot bridle cable. Let the upper segment of the autopilot bridle cable remain on top of the autopilot sector assembly.
4) Remove cotter pin (cable guard) holding the autopilot bridle cable to the bottom of the autopilot sector assembly.
5) Remove the bracket (cable guide) in order to separate the cable assembly from aircraft. (See “Figure 16” on page 221051.)
6) Remove cotter pin, clevis pin, washers (2), and autopilot bridle cable fork end from autopilot sector assembly.
7) Remove bolts and washers (4 ea.) securing pitch servo (GSA 8X/GSM 8X) to mounting brackets and remove pitch servo with attached bridle cable.

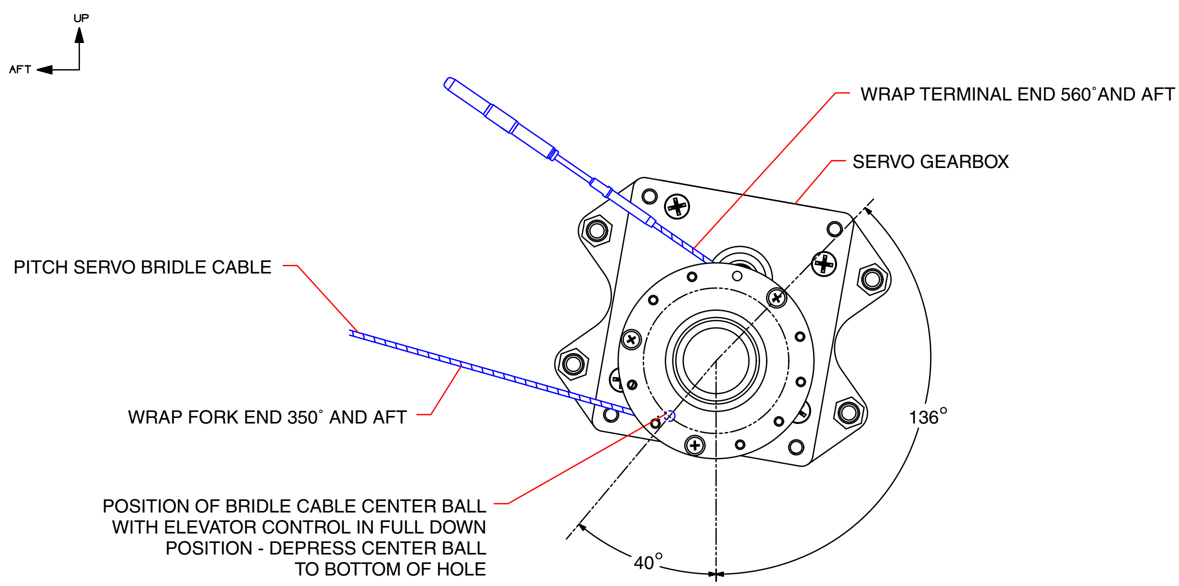
(b) Installation

NOTE: If installing a new, repaired, or exchanged GSM 85A servo gearbox, adjust slip-clutch torque per “Servo Clutch Torque Adjustment” on page 221057 before installation.

1) Rig elevator controls per Elevator Controls Rigging, 27-30-00.
2) Remove screws (3) and remove capstan cover and cable guards from servo.
3) Wrap autopilot bridle cable, align capstan, and press center ball in place per “Figure 18”.
4) Replace cable guards and capstan cover, secure with screws (3). Torque screws per “Figure 13” on page 221044 or “Figure 14” on page 221046.
5) Position pitch servo (GSA 8X/GSM 8X) as shown in “Figure 18” and secure with bolts and washers (4 ea.). Torque bolts per “Figure 18”.
6) Attach both bridal cables to sector assembly with appropriate hardware as shown in “Figure 18”.
7) Install cotter pins (2 ea.) in sector assembly 106588-002 as shown in “Figure 18”.
8) Install and adjust cable guards as shown in “Figure 19”.
9) Connect bridle cable turnbuckle and adjust cable tension as specified in “Figure 18”.
10) Install bridle cable turnbuckle locking clips (2).
11) Connect autopilot harness.
12) Review “Return to Service” on page 221043 under Garmin Servos, to determine if software needs to be reloaded.
13) Check elevator controls for free and correct movement.
14) Check autopilot system for binding and correct operation. See “Ground Checks” on page 221027.
15) Replace the tailcone access panel and secure with twenty-three (23) screws and washers.

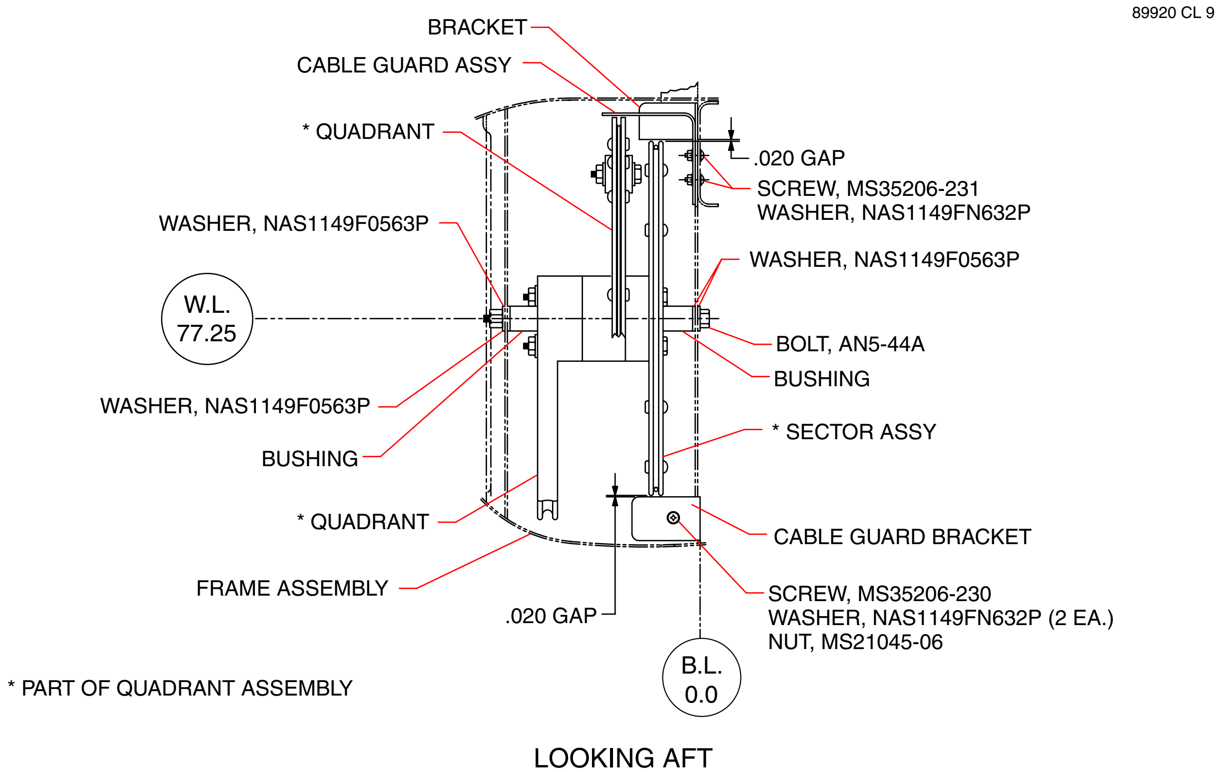


LOOKING LEFT
ELEVATOR SHOWN IN FULL DOWN POSITION



VIEW A
PITCH SERVO BRIDLE CABLE WRAP DETAIL
(ADAPTER AND STRUCTURE REMOVED FOR CLARITY, ENLARGED 50%)

Pitch Servo Installation
Figure 18



Quadrant Assy Setup
Figure 19

F. Servo Clutch Torque Adjustment

(PIR-PPS60236, Rev. D.)

This procedure is written to adjust servo slip clutch torque in GSM 85A Servo Gearboxes. It can also be used to verify servo slip clutch torque in GSM 86 Servo Gearboxes.

NOTE: If verifying GSM 86 slip clutch torque, see “Chart 4” on page 221058. Also, keep in mind these units are set at the factory and cannot be adjusted. Ignore those parts of the procedure that address adjusting to obtain a torque value.

(1) Required Equipment

Garmin Slip-Clutch Fixture, P/N T011-01085-XX

NOTE: Detailed operating instructions for the fixture are in the latest revision of Garmin’s GSA8X / GSM85(A) Installation Manual, P/N 190-00303-72.

(2) Servo Adjustment Fixture Procedure

See “Chart 3” on page 221058 and “Figure 20” on page 221059.

NOTE: To set the torque on Garmin autopilot servos it requires the Garmin Slip-Clutch Fixture P/N T011-01085-XX and for detailed operating instructions for the fixture, refer to the latest revision of Garmin’s GSA8X /GSM85(A) Installation Manual,190-00303-72.

(a) Remove the GSM 85A Servo Mount as per instructions under specific servo, above.

(b) Perform this procedure within a temperature range of 50° F to 120° F.

(c) If a protective cover is installed on the servo mount, remove the protective cover.

(d) Place the servo mount on the slip-clutch adjustment fixture and secure with the toggle clamps. Install the fixture cable between the capstan under test and the fixture capstan. Remove the slack in the cable using the tension adjustment knob. Tighten the two wingnuts on bottom of fixture while holding the top thumbscrews. After wing nuts are tight, deflection using only finger pressure should be no more than 0.25 inch.

(e) Attach a socket to a calibrated torque wrench of appropriate range and place on top of the fixture capstan. Adjust the wrench support fixture so that it contacts the handle in the appropriate location.

(f) Connect a 2 Amp, 24 V, DC power supply to the fixture.

(g) Place the solenoid switch to “ON” and move the direction switch to the clockwise or counterclockwise position. Allow the clutch to rotate at least one revolution in each direction.

(h) Adjust the slip-clutch adjustment (castle) nut on the capstan as needed to obtain the torque value for the given servo listed in “Chart 3”.

NOTE: Position the slip-clutch adjustment (castle) nut to align with holes in the capstan shaft to allow cotter pin installation.

NOTE: During the slip-clutch adjustment of low torque units only (pitch trim), it may be necessary to add an extra washer (AN960-616) under the castle nut used to adjust the slip clutch or remove the existing spacer washer to allow installation of the cotter pin. **DO NOT** use this extra washer if setting a medium or high torque unit.

(i) If the GSM 85A is a low torque unit (pitch trim), refer to the steps below before inserting the cotter pin.

1) Before installing the cotter pin, note how much of the cotter pin hole diameter is above the castle nut.

2) If more than half of the cotter pin diameter is above the nut castellation (see “Figure 21” on page 221059), install the extra washer under the nut. If less than half of the cotter pin diameter is above the nut castellation, the extra washer is not needed, proceed with normal slip-clutch adjustment.

CHART 3
SLIP CLUTCH TORQUE SETTINGS

Servo	Pitch Trim	Pitch	Yaw	Roll
Slip Clutch Torque Value Setting	*32 ±1.5 in.-lbs.	52 ±2 in.-lbs.	50 ±2.5 in.-lbs.	65 ±3 in.-lbs.

* When adjusting the Pitch Trim slip-clutch using the Garmin Slip-clutch Fixture the torque vaue shown on the indicator will read 35 ±1.5 in.-lbs. when the correct torque as shown in the chart is achieved.

CHART 4
GSM 86 SLIP CLUTCH TORQUE SPECIFICATIONS

Servo	Clutch Part Number (Piper/Garmin)	Torque (In.-Lbs.)
Pitch	PS50193-55 / 011-02147-10	52
Pitch Trim	PS50193-58 / 011-02147-01	32
Roll	PS50193-56 / 011-02147-11	61
Yaw	PS50193-55 / 011-02147-10	52

(PIR-89920 CO.)

3) To install the extra washer, do the following steps:
   a) Remove only the castle nut. Do not disassemble the clutch or lift the capstan or hub any amount.
   b) (Install the extra washer directly on top of the existing washer.
   c) Put the castle nut back on the shaft, hand tighten the nut to be sure that the slip clutch is not left loose.
   d) Now reset the clutch again per steps above.

(j) Insert the cotter pin (MS24665-283) into the holes in the shaft (through the notches in the castle nut).

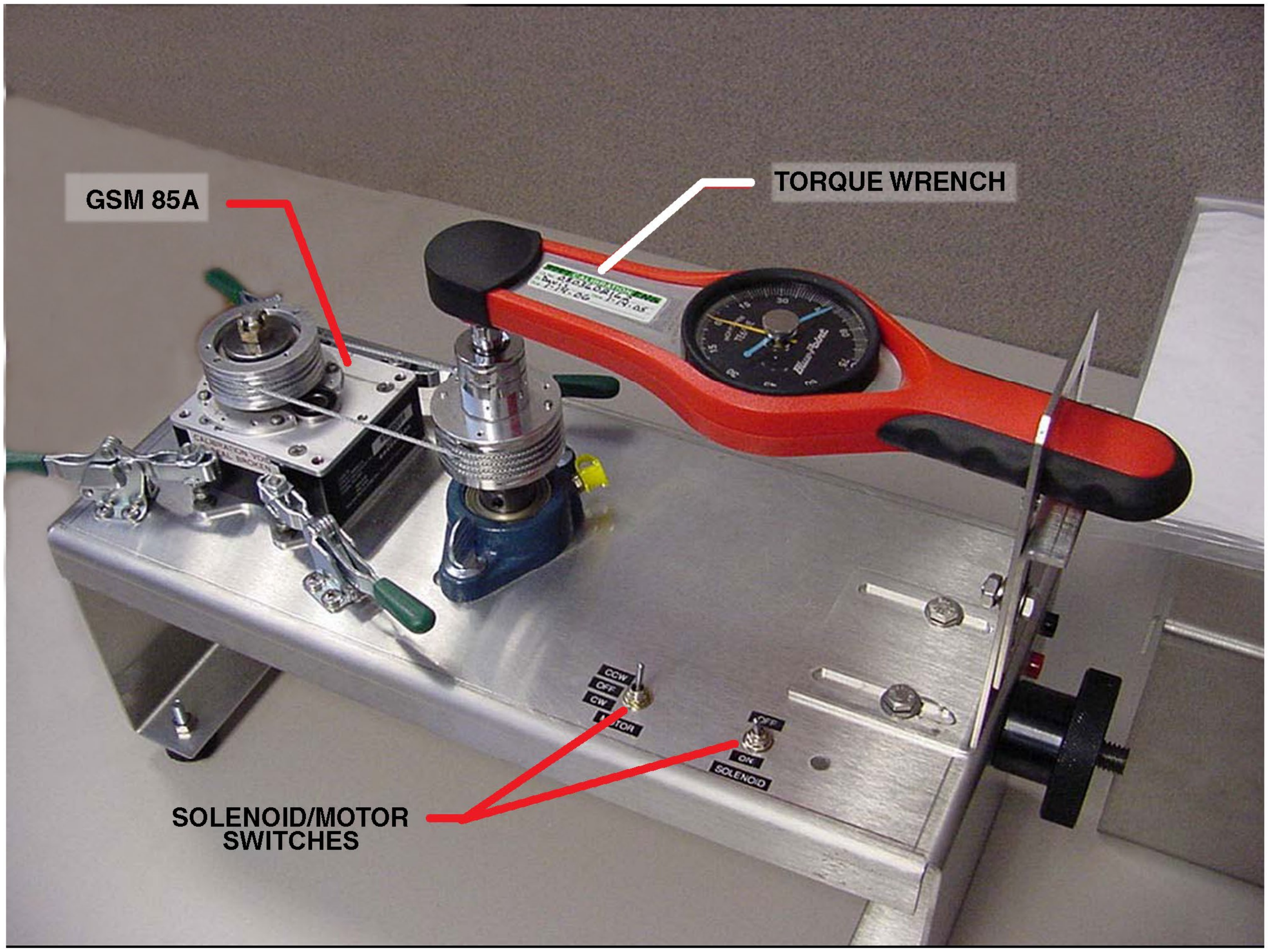
(k) Observe the torque reading for at least two full revolutions in each direction. If the torque reading remains within the required range without any adjustment of the castle nut, proceed to step (12). If not, return to step (7).

(l) When the torque remains within specification, ensure the cotter pin (MS24665-283) is inserted into the holes in the shaft (through the notches in the castle nut). Bend the cotter pin ends to keep the pin in place, using standard cotter pin practices.

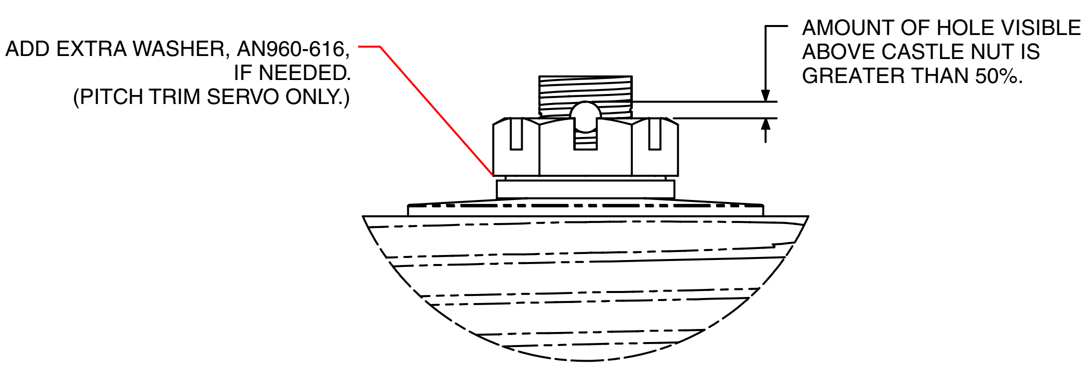
(m) Turn off motor, leaving solenoid on.

(n) Turn motor on in opposite direction (of previous operation). Turn motor off after approx. ½ second, or when tension is relieved in cable (or chain). The tension is relieved when the wrench becomes loose in its support. Also, the torque wrench reading should be zero or close to zero.

(o) Remove torque wrench.



Garmin Slip-Clutch Fixture
Figure 20



Slip-Clutch Castle Nut
Figure 21

(p) Turn off solenoid once tension is relieved.

(q) Turn the power supply off.

(r) When finished, remove the fixture cable, and then remove the servo mount from the fixture.

(s) Install the GSM 85A Servo Mount as per instructions under specific servo, above.

(t) Servo Clutch Torque Adjustment is complete.

CHAPTER

23

COMMUNICATIONS

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CHAPTER 23

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
23-List of Effective Pages	1	Nov 30/17			
	2	Sep 15/09			
23-Table of Contents	1	Nov 30/17			
	2	Sep 15/09			
23-00-00	1	Nov 30/17			
	2	Sep 15/09			
23-10-00	1	Jan 15/13			
	2	Jul 1/10			
	3	Jul 1/10			
	4	Jul 1/10			
	5	Jan 15/13			
	6	Jan 15/13			
	7	Jan 15/13			
	8	Jul 1/10			
	9	Jul 1/10			
	10	Jul 1/10			
23-50-00	1	Nov 30/17			
	2	Nov 30/17			
	3	Nov 30/17			
	4	Nov 30/17			
	5	Nov 30/17			
	6	Nov 30/17			
	7	Nov 30/17			
	8	Nov 30/17			
	9	Nov 30/17			
	10	Nov 30/17			
23-60-00	1	May 31/11			
	2	May 31/11			

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CHAPTER 23 - COMMUNICATIONS

TABLE OF CONTENTS

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GENERAL

1. Radio/Avionics Master Switch

   Avionics Bus #1 and #2 are fed directly from the tie bus, via the avionics master solenoids, which in turn are energized by the radio/avionics master switch. Refer to Chapter 24 and to schematics in 91-24-30.

2. Ground Clearance Switch

   (Optional with conventional instruments or Avidyne installation)

   The ground clearance switch furnishes power to the Comm 1 and speaker panel direct from the battery bus through the ground clearance relay. Refer to 91-23-10 for schematic interface and troubleshooting.

3. Emergency Switch (with G1000/G1000 NXi Installation)

   The EMER switch will activate the #1 COMM/NAV/GPS, pilot's audio panel, along with #1 PFD in reversionary mode with AHRS 1 data, ADC1 data, a subset of engine parameters.

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SPEECH COMMUNICATIONS

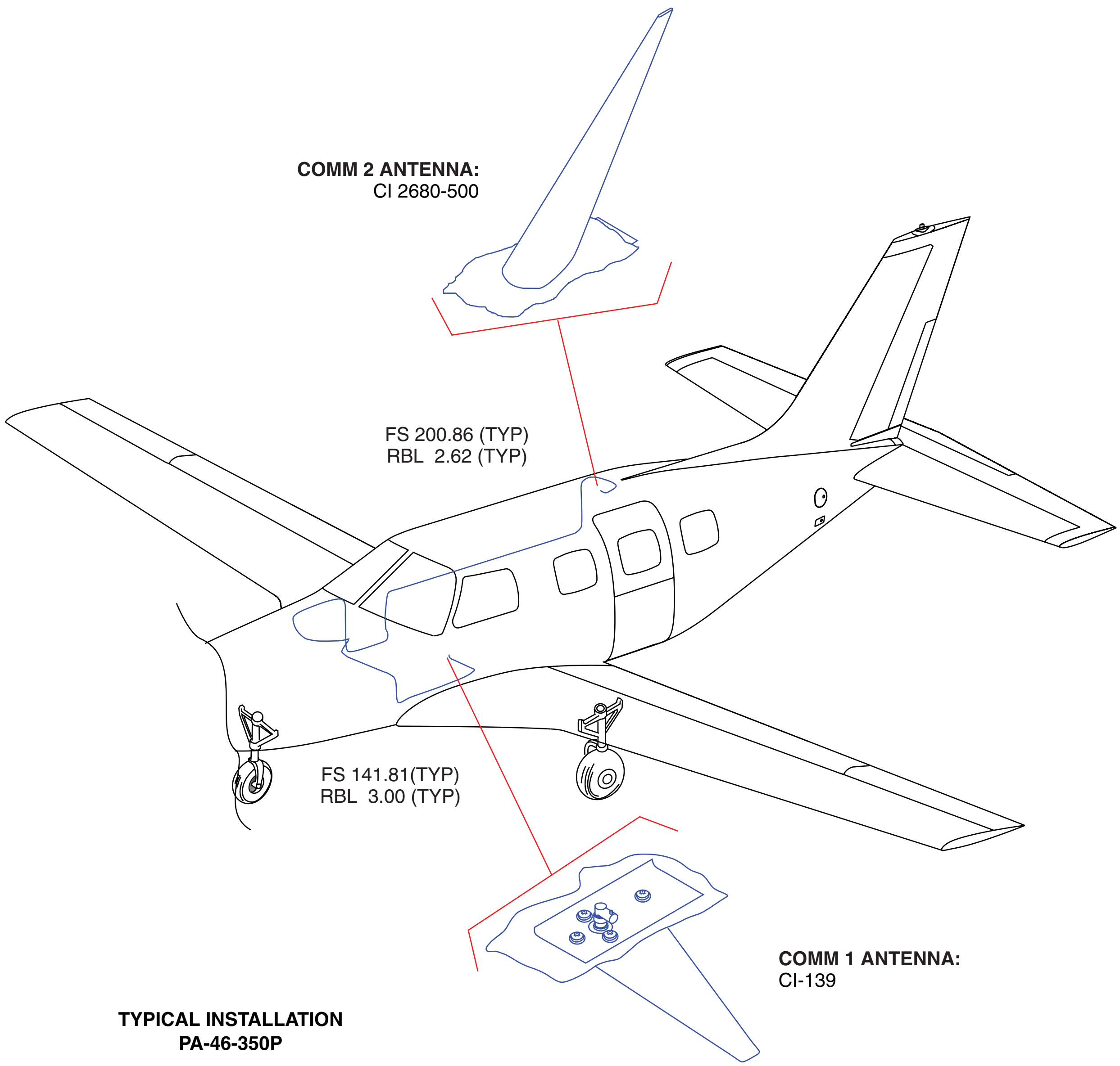
1. COMM 1 (Lower) Antenna Inspection

   NOTE: This inspection incorporates the 4,000 hour repetitive inspection requirements of Piper Service Bulletin No. 1195B. For initial compliance, see the service bulletin.

   In PA-46-350P S/N's 4636375 thru 4636462 (less 4636460): each 4,000 hours time-in-service inspect the COMM 1 (lower) Antenna installation as follows: (see "Figure 1")

   A. Gently try to move the antenna by hand. If the antenna is loose, note the information on the "PA-46 COMM Antenna Skin Inspection Results Form" ("Figure 5").

   B. Inside the aircraft, remove the floor panels above the antenna and retain fasteners for reuse.



Antenna Installation
Figure 1

C. Remove the screws securing the antenna to the aircraft. Note any screws that are loose or damaged. Inspect rivnuts located in the antenna for damage and security. Record condition on the “PA-46 COMM Antenna Skin Inspection Results Form” (“Figure 5”). Do not reuse fasteners, but retain for return to Piper Aircraft, in accordance with Reporting, below.

NOTE: Carefully remove fillet sealant and lower the antenna by the free cable after removing the screws, to prevent possible impact damage from hangar floor.

D. Disconnect cable from the antenna. Remove any gasket that is present between the aircraft skin and the antenna base. Return the gasket to Piper for examination.

CAUTION: AVOID ABRASIVE CLEANING METHODS, SUCH AS SCOTCH BRIGHT AND MECHANICAL MEANS OF CLEANING, WHICH COULD HIDE THE EXISTENCE OF CRACKS.

E. Mask area to prevent damage to paint. Use acetone or alcohol to clean all aircraft skin and the doubler surfaces.

F. Visually inspect aircraft skin and doubler for cracks or scratches. Pay careful attention to all penetrations, rivet holes and the doubler flange. Repair scratches in aircraft skin per Skin Scratch Damage Repair, below.

(1) If aircraft skin is cracked, proceed to Step G.

(2) If aircraft skin is undamaged, but the doubler is cracked or damaged, drill out rivets common to doubler and aircraft outer skin, taking special care to remove rivets without damaging holes. Remove doubler and proceed to Step I.

(3) If aircraft skin and doubler are both undamaged, proceed to Step J, below, and reinstall the antenna.

G. If crack(s) are found in the aircraft skin, determine if they can be removed by enlarging the antenna mounting hole(s) to 3/8 inch (0.375 inch) diameter. If any crack length exceeds this repair limitation, stop all work and contact Piper Aircraft at (772) 299-2141.

CAUTION: SKIN CRACK REMOVAL PROCEDURE DESCRIBED BELOW IS APPLICABLE ONLY FOR THE LOWER ANTENNA MOUNTING HOLES. ANY CRACKS FOUND IN THE UPPER ANTENNA MOUNTING HOLES MUST BE REPORTED TO PIPER AIRCRAFT IN ACCORDANCE WITH REPORTING, BELOW.

H. If all cracks in aircraft skin are within repair limitations defined in Step G, above; remove cracks as follows:

(1) Drill out rivets common to doubler and aircraft outer skin, taking special care to remove rivets without damaging holes. Remove doubler.

(2) Stop drill the crack tip(s) using a #50 drill bit.

(3) Remove crack(s) in aircraft skin by enlarging antenna mounting hole(s) to 0.375 inch diameter using a step drill.

(4) Inspect all open holes to confirm that they are round, straight and free from cracks.

I. If doubler is in serviceable condition, reinstall as described below. If doubler is cracked or otherwise damaged, order and install replacement doubler, P/N 102328-003, as described below:

NOTE: Electrical bonding of the antenna to the aircraft skin is best accomplished by direct metal-to-metal contact of the antenna base to the skin. A resistance of no more that 0.003 ohms between the antenna doubler plate and aircraft skin should be achieved. An approved milliohm meter, calibrated once a year or when questionable, must be used to measure resistance values.

(1) Measure all open rivet holes.

(a) If the hole diameter measures 0.133 inches or less, install replacement rivet of same diameter, MS20426AD4.

(b) If the hole diameter exceeds 0.133 inches, enlarge hole to accept 1/64 oversize replacement rivet. Enlarge hole using number 27 drill bit (nominal diameter 0.144 inches). Do not modify or enlarge existing countersink in skin. Finished hole must be round, straight and free from cracks. Acceptable hole size after drilling is 0.143 to 0.149 inches in diameter. Install NAS1241AD4 rivet.

(2) Shave countersink (manufactured) head flush to within 0.001 to 0.003 inches from outer skin surface.

CAUTION: DO NOT PAINT AIRCRAFT SKIN BELOW ANTENNA.

(3) Touch-up bare aluminum surfaces with suitable primer and topcoat of paint, as required.

J. Reinstall antenna.

(1) Position Spacer, P/N 106774-002, between antenna base and aircraft skin. This spacer is to be used in place of any spacer or gasket that was originally installed from the factory. Replace screws originally installed from the factory with AN3C-4A bolts (P/N 403-051); install new MS35335-32 star washers (P/N 408-100) under head of bolt; add NAS1149F0332P flat washer (P/N 690-609), as shown in “Figure 2”.

(2) If any mounting hole in skin has been enlarged to remove a crack according to instructions in Step H, above, insert Filler Plug P/N 106780-002 into enlarged hole in skin and replace NAS1149F0332P flat washer (P/N 690-609) with replacement AN970-3 flat washer (P/N 407-800), as shown in “Figure 3”.

NOTE: Star washer must not come in contact with the aircraft skin or doubler.

K. Torque bolts to 21 ± 2 inch-pounds.

L. Perform operational test of the COMM system as follows:

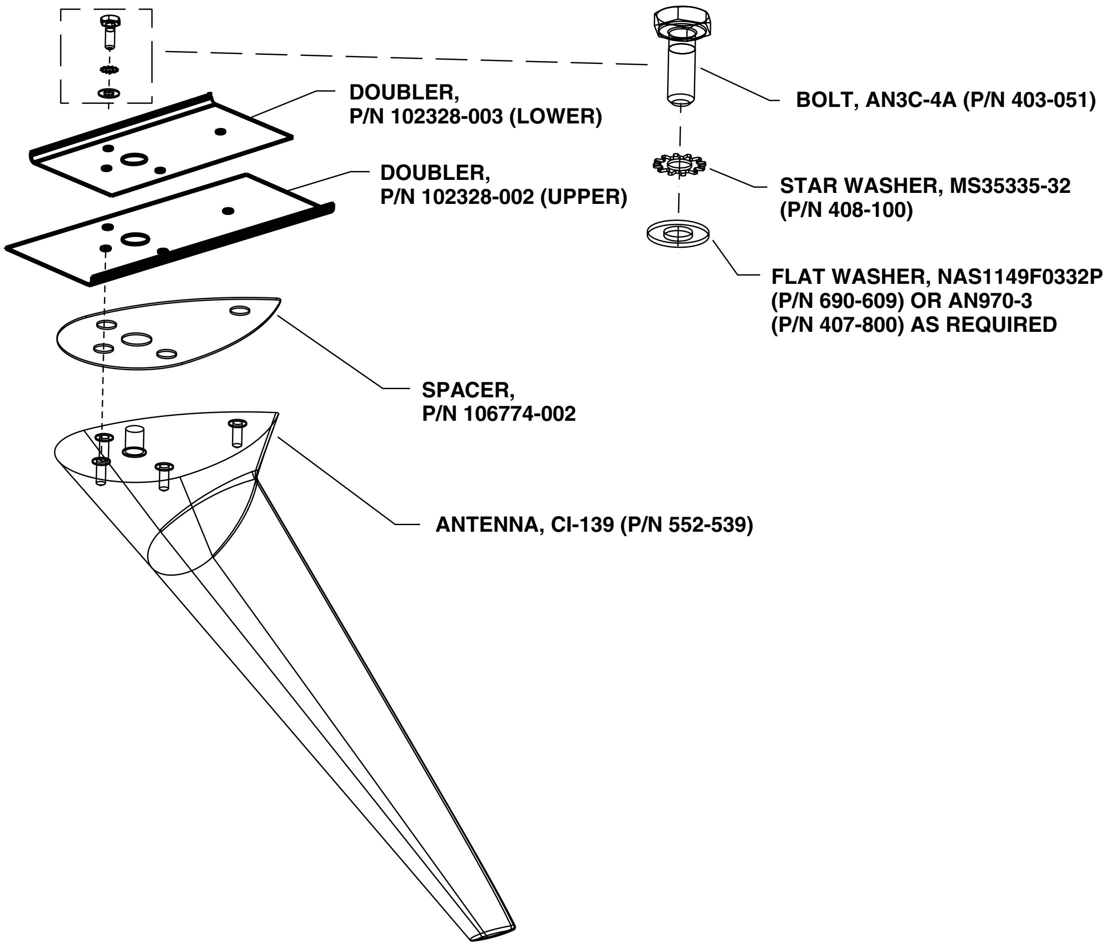
(1) Verify Communication Transceiver operation by exercising the installed transceivers, microphone, microphone key, and audio over the headphones and speaker.

(2) Establish two way communications, and verify that the audio is loud and clear.

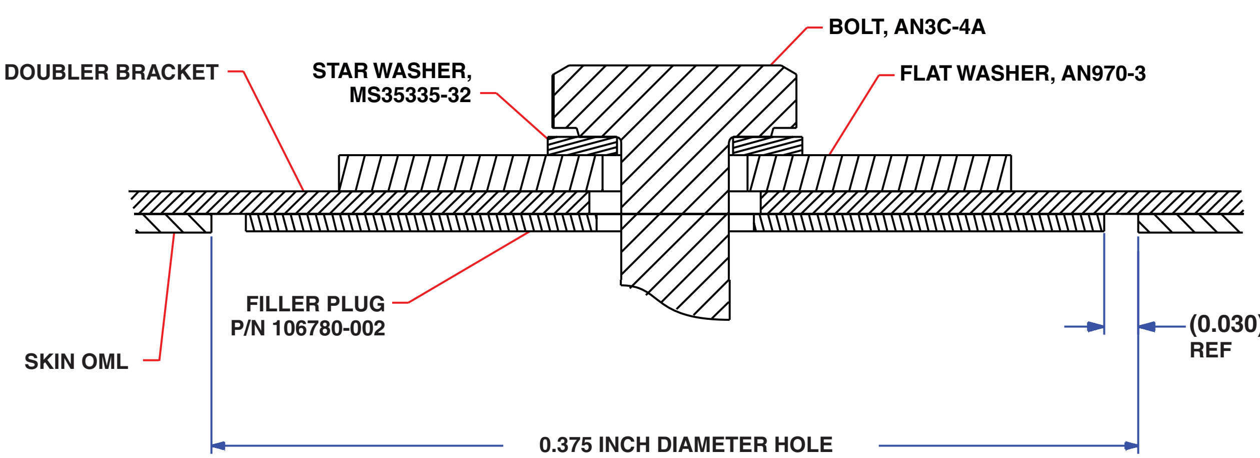
M. Fillet seal around antenna base. Do not seal between antenna, spacer or skin. Use only Bostik 1100FS or 3M Marine Sealant 101. Either sealant can be purchased locally or may be ordered from Aviall: P/N 279-058 (Bostik) or P/N 279-177 (3M Marine Sealant 101).

N. Reinstall floor panels.

O. Make a logbook entry indicating compliance with this inspection.



Antenna Installation - CI-139
Figure 2



Antenna Installation - Hole Plug
Figure 3

23104

2. COMM 2 (Upper) Antenna Inspection

NOTE: This inspection incorporates the 9,400 hour repetitive inspection requirements of Piper Service Bulletin No. 1195B. For initial compliance, see the service bulletin.

In PA-46-350P S/N's 4636375 thru 4636462, less 4636460: each 9,400 hours time-in-service inspect the COMM 2 (upper) Antenna installation per the instructions below.

A. Identify the upper COMM antenna (see "Figure 1") and gently try to move the antenna by hand. If the antenna is loose, note the information on the "PA-46 COMM Antenna Skin Inspection Results Form" (see "Figure 5").

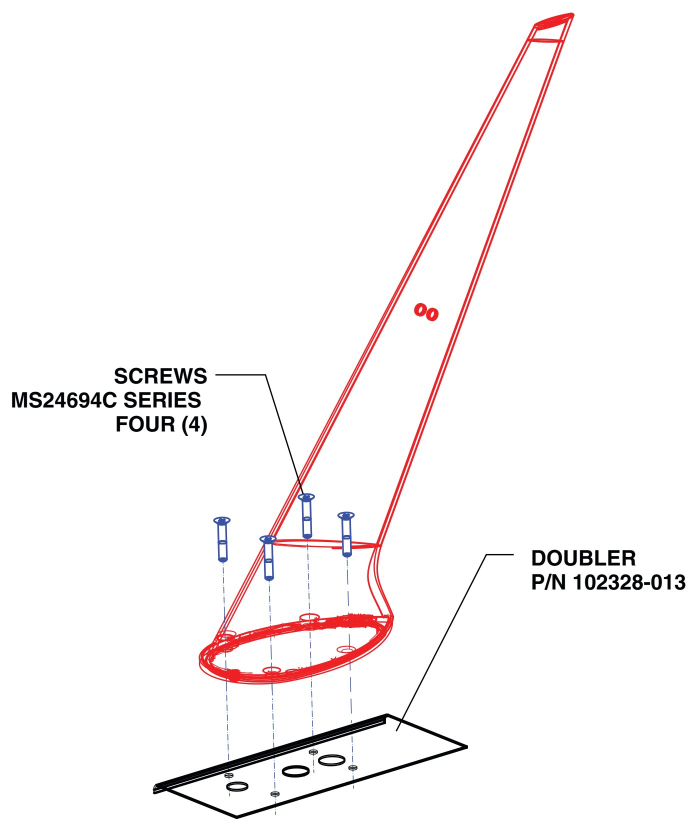
B. Inside the aircraft, remove head-liner to gain access to the mounting area of the upper COMM antenna.

C. Access screw heads of the CI 2680-500 antenna (P/N 683-737 (with Entegra)) from outside the aircraft through counterbored holes in antenna base. See "Figure 4".

D. Remove the screws securing the antenna to the aircraft.

E. Note any screws that are loose or damaged. Do not reuse fasteners, but retain for return to Piper Aircraft, in accordance with "Reporting", below.

NOTE: Carefully remove fillet sealant.



Antenna Installation - CI 2680-500
Figure 4

F. Disconnect the coax cable from the antenna and remove antenna. Remove any gasket that is present between the aircraft skin and the antenna base. Return the gasket to Piper for examination.

CAUTION: AVOID ABRASIVE CLEANING METHODS, SUCH AS SCOTCH BRIGHT AND MECHANICAL MEANS OF CLEANING, WHICH COULD HIDE THE EXISTENCE OF CRACKS.

G. Mask area to prevent damage to paint. Use acetone or alcohol to clean all aircraft skin and doubler surfaces.

H. Visually inspect aircraft skin and doubler for cracks and scratches. Pay careful attention to all penetrations, rivet holes, and the doubler flange. Repair scratches in aircraft skin per instructions in “Skin Scratch Damage Repair”, below.

(1) If aircraft skin is cracked, stop all work and contact Piper Aircraft at (772) 299-2141.

(2) If aircraft skin is undamaged, but the doubler is cracked or damaged, drill out rivets common to doubler and aircraft outer skin, taking special care to remove rivets without damaging holes. Remove doubler and proceed to Step I.

(3) If aircraft skin and doubler are both undamaged, proceed to Step J and reinstall the antenna.

I. If doubler is cracked or otherwise damaged, order replacement doubler, P/N 102328-013 and install as described below:

NOTE: Electrical bonding of the antenna to the aircraft skin is best accomplished by direct metal-to-metal contact of the antenna base to the skin. A resistance of no more that 0.003 ohms between the antenna doubler plate and aircraft skin should be achieved. An approved milliohm meter, calibrated once a year or when questionable, must be used to measure resistance values.

(1) Measure all open rivet holes.

(a) If the hole diameter measures 0.133 inches or less, install replacement rivet of same diameter, MS20426AD4.

(b) If the hole diameter exceeds 0.133 inches, enlarge hole to accept 1/64 oversize replacement rivet. Enlarge hole using # 27 drill bit (nominal diameter 0.144 inches). Do not modify or enlarge existing countersink in skin. Finished hole must be round, straight and free from cracks. Acceptable hole size after drilling is 0.143 to 0.149 inches in diameter. Install NAS1241AD4 rivet.

(2) Shave countersink (manufactured) head flush to within 0.001 to 0.003 inches from outer skin surface.

CAUTION: DO NOT PAINT AIRCRAFT SKIN BELOW ANTENNA.

(3) Touch-up bare aluminum surfaces with suitable primer and topcoat of paint, as required.

J. Reinstall antenna. (See “Figure 4”.)

The nominal screw required is P/N 410-150 (MS24694C57), four places. However, actual required screw length may vary due to tolerance build-up of antenna base, aircraft skin, doubler, and paint thickness. Choose the correct fastener according to the following method:

Begin with the nominal length fastener MS24694C57. With the antenna removed from the aircraft, place the screw into the mounting hole in the antenna base, and measure the unthreaded portion that visibly protrudes through the bottom of the base. This measurement cannot be less than the measured combined thickness of the skin and doubler (as measured using a dial caliper or other suitable tool). Since it is critical that the threaded portion of the installed screw does not make contact with the inside surface of the hole in the skin or doubler, choose the longest screw that can be installed without bottoming the threads in the nutplate.

NOTE: Torque should build up gradually during fastener installation. An abrupt buildup indicates threads are bottoming in the nutplate. If this occurs, choose the next size shorter length screw.

Screw lengths are available from shortest to longest: P/N 410-064 (MS24694C54), P/N 410-026 (MS24694C55), P/N 410-028 (MS24694C56), P/N 410-150 (MS24694C57), P/N 410-073 (MS24694C58), P/N 414-521 (MS24694C59) and P/N 408-839 (MS24694C60).

K. Torque screws/bolts to 21 ± 2 inch-pounds.

L. Perform operational test of the COMM system as follows:

(1) Verify Communication Transceiver operation by exercising the installed transceivers, microphone, microphone key, and audio over the headphones and speaker.

(2) Establish two way communications, and verify that the audio is loud and clear.

M. Fillet seal around antenna base. Do not seal between antenna, spacer or skin. Use only Bostik 1100FS or 3M Marine Sealant 101. Either sealant can be purchased locally or may be ordered from Aviall: P/N 279-058 (Bostik) or P/N 279-177 (3M Marine Sealant 101).

N. Reinstall head-liner.

O. Make a logbook entry indicating compliance with this inspection.

3. Skin Scratch Damage Repair

Scratch damage may be repaired by burnishing scratches manually to a smooth nick free finish at the edge of each hole. Use # 400 silicon carbide paper or its equivalent when refinishing scratch damage. Do not use abrasives containing iron bearing materials on aluminum alloys.

A. Limitations

The reworked area shall not exceed 1/2 inch to either side of the damage. The rework depth shall be sufficient to remove evidence of the scratch, within the following limitations:

Material Nominal Thickness	Minimum Material Thickness After Rework
0.032 (upper)	0.0290 (upper)
0.040 (lower)	0.0365 (lower)

NOTE: If the minimum thickness of the aircraft skin is less than 0.0290 (upper) or 0.0365 (lower), stop work and contact Piper Aircraft at (772) 299-2141.

B. Return to Step F in the COMM 1 (Lower) Antenna Inspection or Step H in the COMM 2 (Upper) Antenna Inspection, as applicable.

4. Reporting

If any cracks are found during any inspection, complete and return "PA-46 COMM Antenna Skin Inspection Results Form" (Figure 5) along with pictures or sketches of all crack(s) to Piper Aircraft. Send form per instructions on form.

Also return original hardware to Piper for evaluation. Send hardware to:

Piper Aircraft, Inc.
2926 Piper Drive
Vero Beach, FL 32960
Attention: Customer Service Department

PA-46 Comm. Antennas Skin Inspection Results Form

Aircraft Serial Number:________________________

Aircraft Registration No.: ____________________________

Owner Name:____________________________________

Address:____

Phone Number:__________________________________

E-mail address:__________________________________

Total Time on A/C:_______________________________

Inspection revealed:

Loose condition.	☐	Upper	☐	Lower	☐
No cracks found.	☐	Upper	☐	Lower	☐
Crack(s) found.	☐	Upper	☐	Lower	☐

Notes: ____
____
____
____

This information can be forwarded to Piper via:

E-mail: airframe@piper.com

FAX: (772) 978-6573

Phone No.: (772) 299-2325

PA-46 COMM Antenna Skin Inspection Results Form
Figure 5

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AUDIO INTEGRATING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Garmin GMA 340 Audio Control Panel

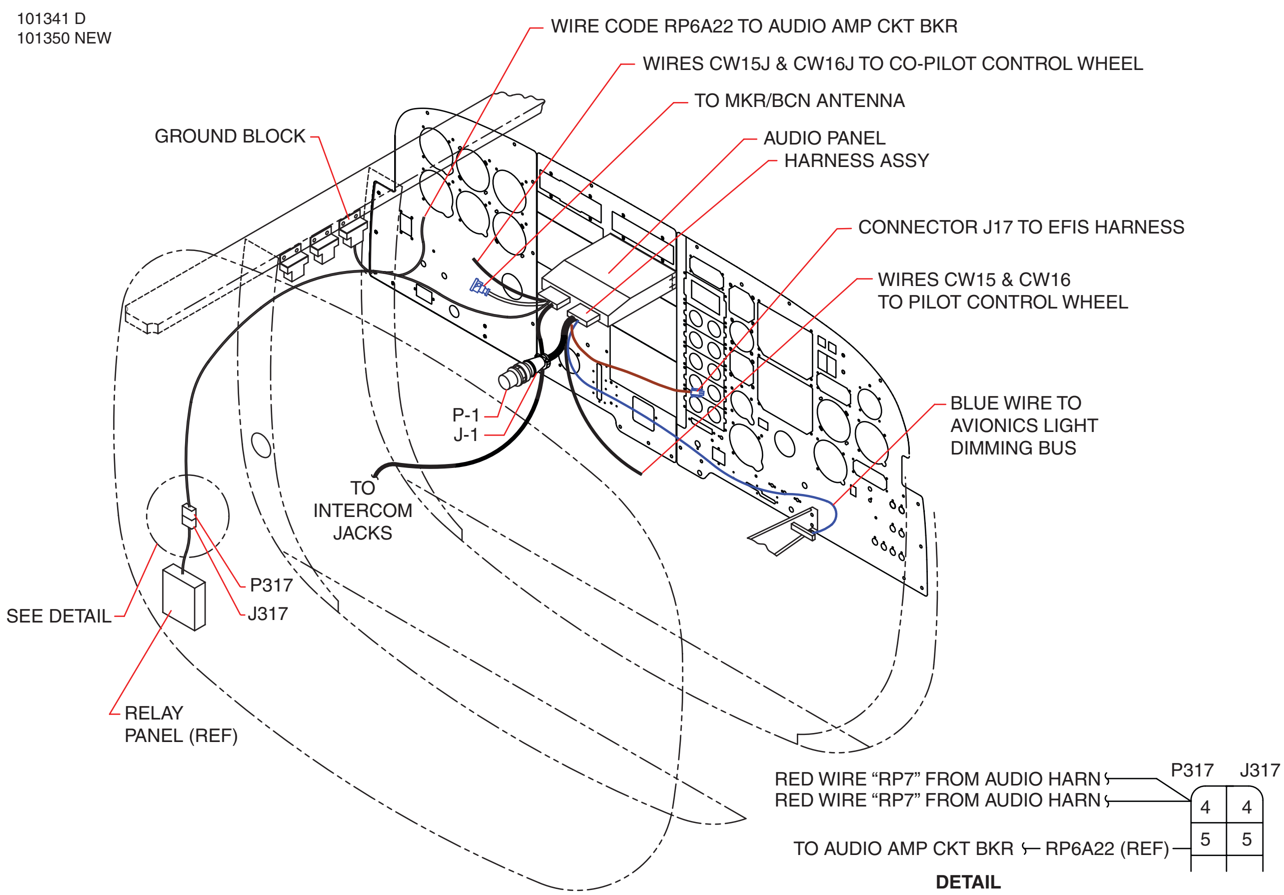
See “Figure 1”.

The Garmin GMA 340 Audio Panel is installed as standard equipment in PA-46-350P S/N’s 4636248–4636459, 4636461–4636462, and 4636481; and PA-46R-350T S/N’s 4692001–4692133, 4692141, 4692149 and 4692153. It integrates the audio switching, amplifier, and intercom system with a marker beacon receiver. Maintenance of the GMA 340 is “on condition” only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop, in accordance with the GMA 340 Audio Panel Maintenance Manual (Garmin P/N 190-00149-02).

Information provided in this manual is intended to aid the removal and installation of the GMA 340 unit and its associated wiring, and to permit basic system functional test and adjustment. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See 39-10-00 - Rack-Mounted Avionics, Removal and Installation.



GMA-340 Audio Panel Installation (Typical)
Figure 1

B. System Functional Test

See “Figure 2” on page 23503.

(1) On Ground

The following known good equipment is required prior to performing this test: microphone, headset, speaker and avionics receivers.

(a) Lamp Test

1 Apply power to the unit by rotating the pilot intercom knob clockwise.

2 The test button checks the internal LED annunciator and marker beacon lamps. Press TEST to confirm operation of the LED’s. Cover the photocell with a finger and observe that the LED annunciators dim automatically. Check the front panel back-lighting and dimming function. Each annunciator contains a lamp for illumination.

(b) Fail-safe Operation Check

1 Turn the unit off by rotating the pilot intercom knob counterclockwise.

2 Check fail-safe operation by exercising the COM 1 microphone, microphone key and audio over the headphones.

3 Turn the unit back on to continue testing.

(c) Transceiver Operational Check

1 Perform a ramp test radio check by exercising the installed transceivers, microphone, microphone key and audio over the headphones and speaker.

2 Verify that communications are loud and clear and push-to-talk (PTT) operation is correct.

(d) Intercom System (ICS) Check

1 Set the intercom to the ALL mode (Crew and Pilot LED’s off).

2 Plug in headsets at each ICS position.

3 Adjust squelch and volume for each position and verify that the ICS is working properly.

4 Check Pilot and Copilot ICS positions for isolation and proper operation of volume and squelch controls.

5 Press the PA button. Verify that microphone audio is heard over the speaker.

(e) Aircraft Receivers Check

1 Select the audio source corresponding to each installed avionics unit and check for audio over the headsets.

2 Check for Pilot/Copilot audio isolation when pressing the COM 1/2 button.

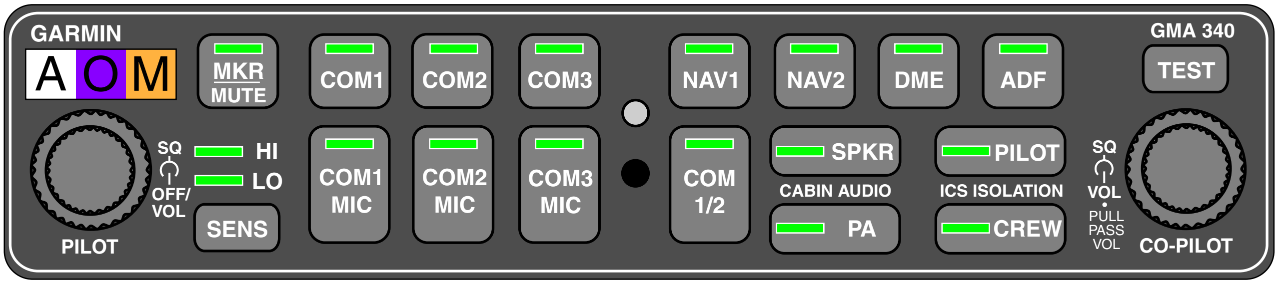
3 Press the SPKR button and verify that any selected audio is heard over the speaker.

(f) Music System Check

Connect a stereo music source to MUSIC 2. Press the CREW button to set the ICS to the crew mode. Verify that stereo audio is heard in the passenger headsets only.

(2) In Flight

Verify proper operation of the marker lamps and marker audio, including the marker audio mute function. Check proper operation of the marker sensitivity selection (using the SENS button) by flying towards the outer marker position initially using HI sensitivity. When the OM audio is just barely audible in the headset, switching to LO sensitivity should reduce or eliminate the audio.



GMA 340 Audio Panel
Figure 2

C. Adjustment

See "Figure 3" on page 23504.

CAUTION: USE ONLY A 2 MM (MAX BLADE WIDTH) FLAT-BLADE NON-CONDUCTIVE SCREW DRIVER AS AN ADJUSTMENT TOOL. BE CAREFUL WHEN INSERTING ADJUSTMENT TOOL THROUGH THE TOP COVER. THE UNIT MAY BE DAMAGED IF AN ADJUSTMENT TOOL IS ACCIDENTALLY FORCED AGAINST UNINTENDED COMPONENTS OR CIRCUIT BOARD PATHS.

The following adjustments can be made through access holes in the top cover of the GMA 340:

(1) Marker beacon audio level.

Counter-clockwise adjustment increases the marker audio level.

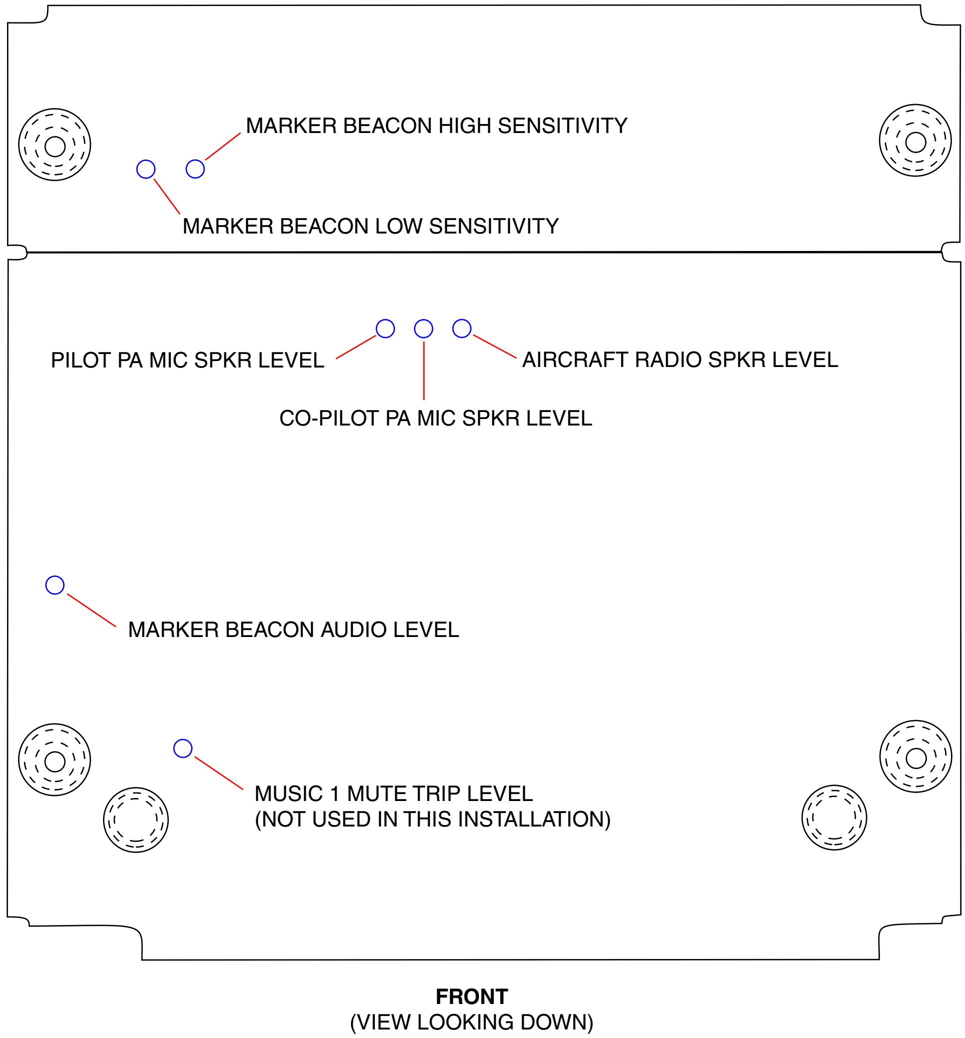
(2) Marker beacon sensitivity.

Clockwise (CW) adjustment increases the sensitivity. LOW sensitivity can be adjusted without affecting HIGH sensitivity setting. Adjusting HIGH sensitivity will, however, affect the LOW sensitivity. If the HIGH sensitivity setting is adjusted, then the LOW sensitivity setting should be checked and adjusted afterwards, as needed. If your GMA 340 top cover does not have the marker beacon sensitivity adjustment access holes as indicated in "Figure 3", and you need to adjust the sensitivity, contact Garmin for instructions.

(3) Aircraft radio speaker output level.

(4) Pilot PA microphone speaker output level.

(5) Copilot PA microphone speaker output level.



GMA 340 Audio Panel Adjustments
Figure 3

2. M1091 Summing Amplifier (with Avidyne EFIS only)

(PIR-PPS60205, Rev. A.)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The Baker Electronics M1091 Summing Amplifier is mounted on the underside of an access cover in the cabin floor, just aft of F.S. 100.00, on the pilot's side. The Summing Amp. is connected to the GMA-340 Audio Panel, the audio switching amplifier, and intercom system. It converts warning tones from up to six audio channels into one audio output. Maintenance is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop.

Information provided herein is intended to aid the removal and installation of the M1091 Summing Amp. and its associated wiring and to permit basic system functional test and adjustment. In addition, an electrical schematic is provided in 91-23-50.

When replacing the Summing Amplifier Assy. the only adjustment required is to ensure that any unused channels are turned off. Proceed as follows:

A. Setup

(1) Remove the four #4-40 x 3/8 pan head phillips screws and four #4 lockwashers located on the sides of the M1091.

(2) Remove two D-Subminiature connector male screw locks located on either side of the connector.

(3) Remove the case of the M1091.

(a) Remove the case by tilting the side of the case opposite the connector up at a 30 degree or greater angle.

(b) Slide the case toward the connector and remove it.

B. Adjustment

See "Chart 1" and "Figure 4" on page 23506.

(1) Input channels not used on the M1091 will be turned OFF by adjusting the corresponding channel potentiometer fully counter clockwise.

(2) Input channels that are used on the M1091 will be turned on by adjusting the corresponding channel potentiometer fully clockwise.

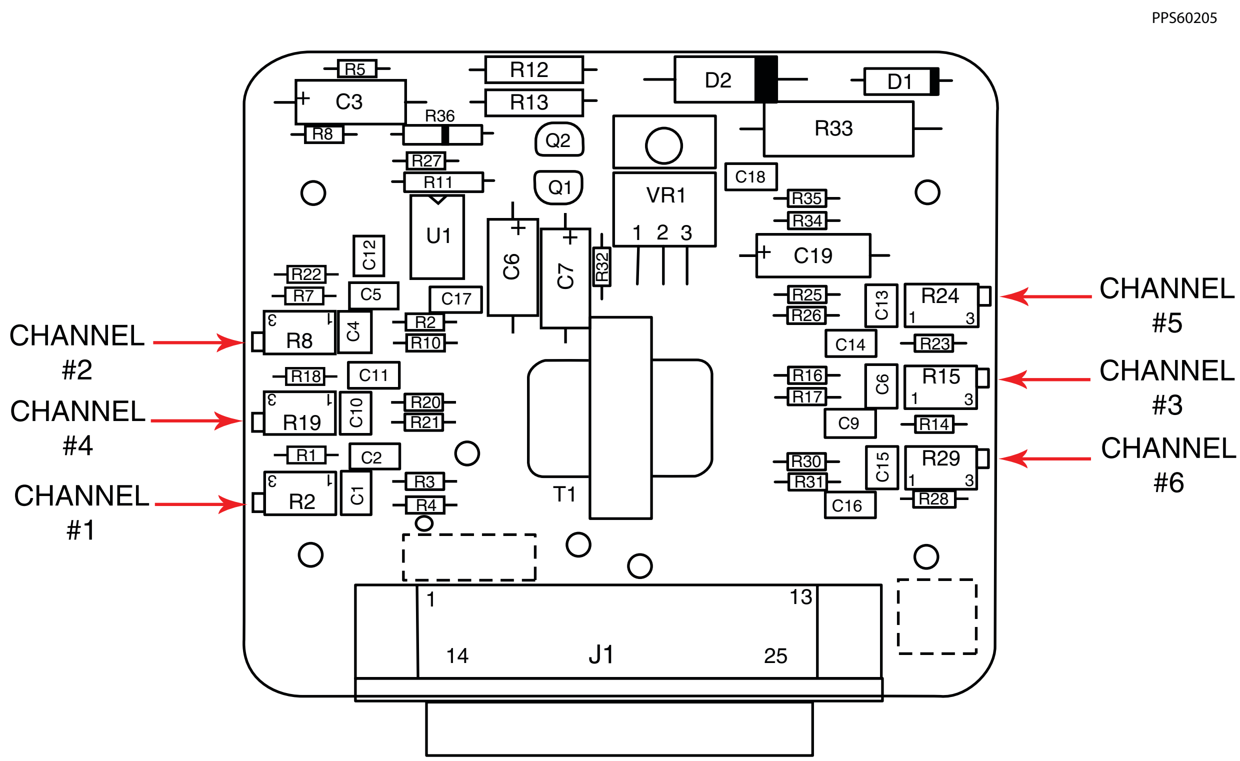
(3) After all channels have been adjusted, reinstall the unit case and the case and connector screws previously removed.

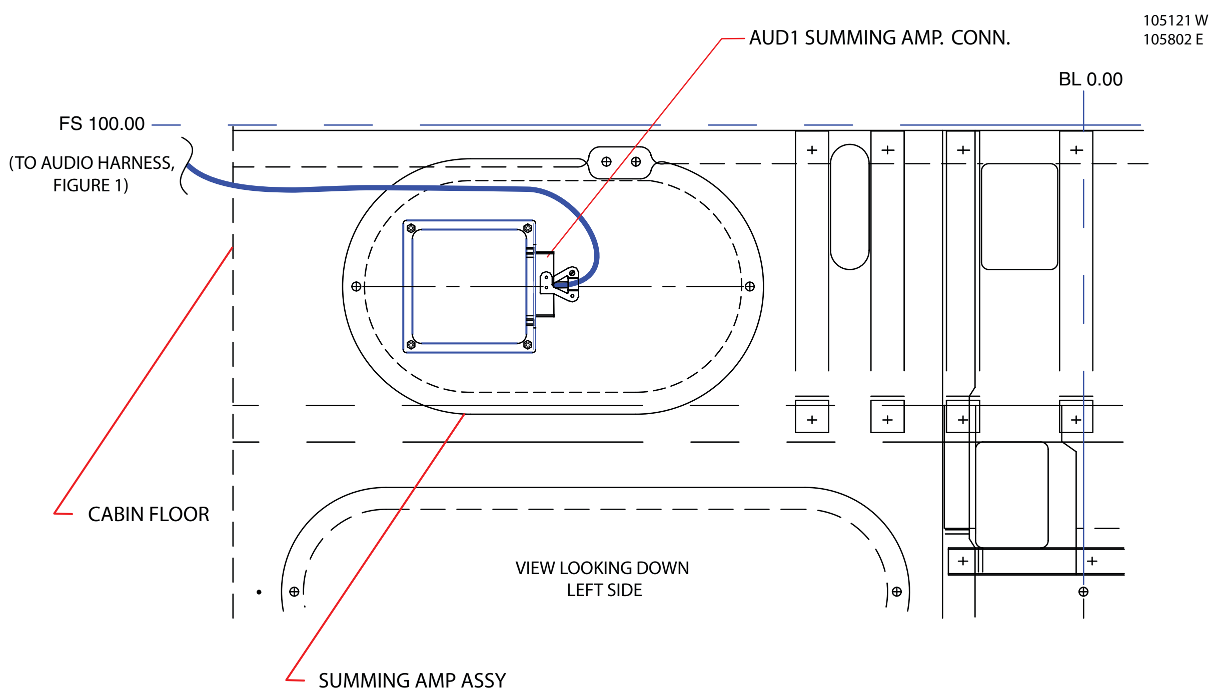
C. System Test

Activate each audio source and verify that the desired audio output is produced.

CHART 1
M1091 SUMMING AMPLIFIER CONFIG PAGE

	AUDIO SOURCE	AUDIO SOURCE
MODEL	PA-46-350P	PA-46R-350T
CHANNEL 1	#1 MODE S TRANSPONDER	TAS 610
CHANNEL 2	#2 MODE S TRANSPONDER	AUTOPILOT
CHANNEL 3	RADAR ALTIMETER	TRIM MONITOR
CHANNEL 4	IHAS	UNUSED
CHANNEL 5	TAWS	UNUSED
CHANNEL 6	UNUSED	UNUSED





M1091 Summing Amplifier
Figure 4

3. Garmin GMA 347 Audio Control Panel

The GMA 347 is installed as standard equipment S/N's 4636460, 4636463–4636651, less 4636481 and 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

It integrates NAV/COM digital audio, telephone, intercom system and marker beacon controls. The Audio Panel communicates with both GIA 63Ws using RS-232 digital interface. The GIA 63W contains the GPS receiver, VHF COM/NAV receivers. A fail-safe circuit connects the pilot's headset and microphone directly to COM 1 in case power is interrupted or the unit is turned off.

Maintenance of the GMA 347 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop. See 34-25-01.

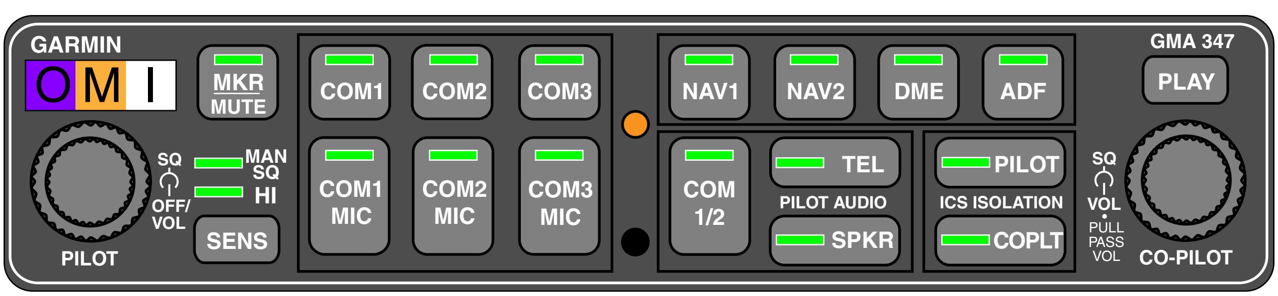
Information provided in this manual is intended to aid the removal and installation of the GMA 347 unit and its associated wiring and to permit basic system functional test. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See GMA 347 under Components in 34-25-01.

B. System Functional Test

See GMA 347 Test under LRU Test Procedures in 34-25-01.



GMA 347 Audio Panel
Figure 5

4. Garmin GMA 350 Audio Control Panel

The GMA 350 is installed as standard equipment in S/N's 4636633, 4636652–4636715, 4636717–4636719.

It integrates NAV/COM digital audio, telephone, intercom system and marker beacon controls. The Audio Panel communicates with both GIA 63Ws using RS-232 digital interface. The GIA 63W contains the GPS receiver, VHF COM/NAV receivers. A fail-safe circuit connects the pilot's headset and microphone directly to COM 1 in case power is interrupted or the unit is turned off.

Maintenance of the GMA 350 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop. See 34-25-01.

Information provided in this manual is intended to aid the removal and installation of the GMA 350 unit and its associated wiring and to permit basic system functional test. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See GMA 350 under Components in 34-25-01.

B. System Functional Test

See GMA 350 Test under LRU Test Procedures in 34-25-01.

GMA 350 Audio Panel
Figure 6

5. Garmin GMA 350c Audio Control Panel

The GMA 350c is installed as standard equipment in S/N's 4636716, 4636720 and up.

It integrates NAV/COM digital audio, telephone, intercom system and marker beacon controls. The GMA 350c also has Bluetooth capability and can connect with various Bluetooth devices. The Audio Panel communicates with both GIAs using RS-232 digital interface. The GIA contains the GPS receiver, VHF COM/NAV receivers. A fail-safe circuit connects the pilot's headset and microphone directly to COM 1 in case power is interrupted or the unit is turned off.

Maintenance of the GMA 350c is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop. See 34-25-02.

Information provided in this manual is intended to aid the removal and installation of the GMA 350c unit and its associated wiring and to permit basic system functional test. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See GMA 350c under Components in 34-25-02.

B. System Functional Test

See GMA 350c Test under LRU Test Procedures in 34-25-02.

GMA 350c Audio Panel
Figure 7

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STATIC DISCHARGING

1. Static Wicks

A. Description

This airplane is equipped with 16 static wicks (dischargers): three on the outboard end of each aileron, three on the outboard half of each side of the elevator; two on the right-hand side of the rudder and one on the left-hand side of the rudder, and one in the center of the elevator trim tab.

B. Inspection

Each 100 hours or annually, whichever comes first, inspect static wicks as follows:

(1) General appearance and physical condition.

(2) Security of attachment to airframe.

(3) Discharge points visible.

(4) Resistance (P/N 456-872) = 1.0 to 200 megohms (500 to 1,000 volt megohmmeter).
Resistance (P/N 456-904) = 6.0 to 200 megohms (500 to 1,000 volt megohmmeter).

(5) Base resistance to airframe (one (1) ohm maximum).

C. Removal

Unscrew static discharger from mounting base. Capture and retain lock washer.

D. Installation

NOTE: Static discharger P/N 456-904 is installed in S/N's 4636491 and up; and 4692158 and up; and authorized service replacement in 4636001 thru 4636490 and 4692001 thru 4692157.

(1) Install/verify lock washer on Static Discharger. If present, discard plastic lockwasher retainer.

NOTE: P/N 456-904 buildup will be as shown in "Figure 1". Earlier dischargers use only a single lock washer.

(2) Install Static Discharger into the mounting base. Ensure snug fit.

(3) Tighten hex nut to mounting base. Torque P/N 456-904 only 5 in.-lb.

(4) Verify electrical resistance between Static Discharger tip and aircraft surface per inspection, above.

E. Mounting Base Installation

NOTE: Mounting base replacement is not routinely required. Static dischargers P/N 456-904 are compatible with the original mounting bases.

(1) Drill out rivets, remove old mounting base and discard.

(2) Using the new Mounting Base as a template, outline pattern on the aircraft surface.

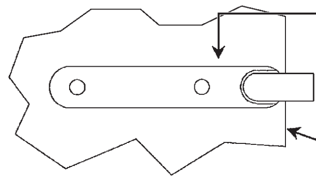
(3) Remove paint/oxides from aircraft surface and on bottom of mounting base using #400 grit sandpaper.

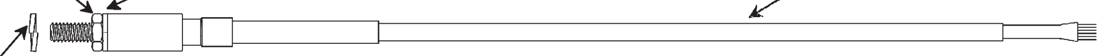
(4) Secure mounting base to surface with rivets.

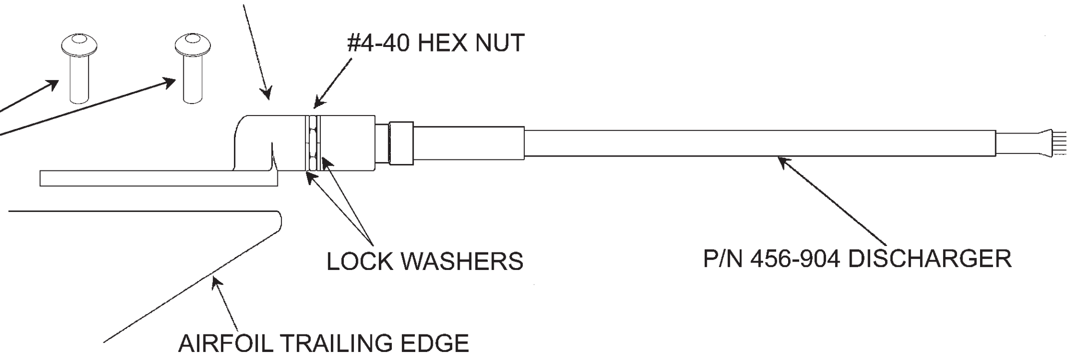
(5) Verify electrical bond between mounting base and aircraft surface. Resistance shall not exceed 0.100 ohms (using low voltage milli-ohm or micro-ohmmeter).

(6) Apply flexible sealant around the mounting base edges and fastener heads to protect against moisture ingress. Fillet all around mounting base edges. (See "Figure 1".)

(7) Allow flexible sealant to cure at least 24 hours at 70° F, before installation of static discharger.







Effectivity
4636001 and up
4692001 and up

P/N 456-904 Static Discharger Installation
Figure 1

2. Bonding Straps

(See also 51-80-00.)

(PIR-PPS55006, Rev. AH.)

To aid in dissipating static electricity buildup, the ailerons, elevator and rudder are bonded to either the control's hinge or spar.

When replacing the jumper assemblies (bonding straps), secure the end of the jumper that mounts to the control's hinge or spar as follows: (see also 51-80-00)

A. Clean an area of 1 1/2 times the diameter of the jumper's washer down to bare metal.

B. Attach the jumper and washer to the control's hinge or spar.

C. Seal the cleaned area with a light coat of epoxy primer within 24 hours.

CHAPTER

24

ELECTRICAL POWER

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CHAPTER 24

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
24-List of Effective Pages	1	Mar 25/21
	2	Mar 25/21
24-Table of Contents	1	Mar 25/21
	2	Mar 25/21
	3	Mar 25/21
	4	Nov 30/17
24-00-00	1	Mar 25/21
	2	Mar 25/21
	3	Nov 30/17
	4	Nov 30/17
	5	Mar 25/21
	6	Mar 25/21
24-20-00	1	Mar 25/21
	2	Nov 30/17
	3	Mar 25/21
	4	Nov 30/17
	5	Nov 30/17
	6	Nov 30/17
	7	Nov 30/17
	8	Mar 25/21
	9	Nov 30/17
	10	Nov 30/17
	11	Nov 30/17
	12	Nov 30/17
	13	Mar 25/21
	14	Mar 25/21
	15	Mar 25/21
	16	Mar 25/21
24-30-00	1	Nov 30/17
	2	Nov 30/17
	3	Nov 30/17
	4	Nov 30/17
	5	Nov 30/17
	6	Nov 30/17
	7	Nov 30/17
	8	Sep 15/09
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	26	Nov 30/17
	27	Nov 30/17
	28	Nov 30/17
	29	Nov 30/17
	30	Nov 30/17
	31	Nov 30/17
	32	Nov 30/17
24-40-00	1	Nov 30/17
	2	Dec 15/14

CHAPTER 24

LIST OF EFFECTIVE PAGES (Continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
24-60-00	1	Nov 30/17			
	2	Mar 25/21			
	3	Mar 25/21			
	4	Mar 25/21			
	5	Mar 25/21			
	6	Mar 25/21			
	7	Mar 25/21			
	8	Mar 25/21			
	9	Mar 25/21			
	10	Mar 25/21			
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	26	Nov 30/17			
	27	Mar 25/21			
	28	Mar 25/21			
	29	Nov 30/17			
	30	Mar 25/21			
	31	Mar 25/21			
	32	Mar 25/21			
	33	Mar 25/21			
	34	Mar 25/21			

CHAPTER 24 - ELECTRICAL POWER

TABLE OF CONTENTS

CHAPTER 24 - ELECTRICAL POWER

TABLE OF CONTENTS (continued)

CHAPTER 24 - ELECTRICAL POWER

TABLE OF CONTENTS (continued)

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GENERAL

This section contains information to assist in servicing the electrical power distribution systems and components. This chapter does not include autopilot or avionics service information. Refer to Chapter 91 for electrical systems schematics.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Description

The electrical system is 28 Vdc, negative ground, consisting of:

A.	Two 70-amp, Prestolite alternators (S/N's 4636001 thru 4636131)	standard, or
	Two 75-amp, Electrosystems alternators (S/N's 4636132 and up) (S/N's 4692001 and up)	standard
B.	Two Lamar Alternator Control Units	standard
C.	One Digital volt meter (See Note below)	standard
D.	Two ammeters (See Note below)	standard
E.	Dual main power distribution bus/tie bus	standard
F.	One 10 amp-hour lead acid battery-manifold type-Gill (S/N's 4636001 thru 4636374)	standard
	One 16 amp-hour lead acid battery-recombinant gas type-Gill (S/N's 4636375 thru 4636515) (S/N's 4692001 thru 4692172)	standard
	One 16 amp-hour lead acid battery-recombinant gas type-Concorde (S/N's 4636516 and up) (S/N's 4692173 and up)	standard
	and (S/N's 4636001 thru 4636515) (S/N's 4692001 thru 4692172)	with Kit P/N 88495-002
G.	One acid recovery jar with venting overboard (S/N's 4636001 thru 4636374)	standard
H.	External power provisions	standard
I.	Visual annunciation with alternator "OUT" sensors	standard

NOTE: Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.

Each of the dual alternators provides 28 Vdc electrical power. Both are negative grounded. The battery is located beneath the floor of the forward baggage compartment, pilot's side. The battery is hard-wired directly to the battery bus. An external power circuitry is provided for ground operational requirements. The manifold-type battery (S/N's 4636001 thru 4636374) is vented overboard through an acid recovery system.

Electrical switches are located as follows:

An overhead switch panel located above the windshield (See 11-30-00, Figure 2).

Avionics and systems switches located on the instrument panel.

Environmental control panel installed in the instrument panel.

On G1000/G1000 NXi equipped aircraft, a battery bus, located in the baggage compartment, provides a continuous source of power for the emergency locator transmitter switch (when equipped with ELT 110 or ME406 ELT only), forward baggage compartment light, and aft cabin courtesy lights. Because the battery bus is connected directly to the battery, power is available for these functions even when the Battery Master switch is OFF. The battery bus contains fuses to protect these circuits.

The EMERGENCY bus (See “Chart 1”) is activated by depressing the EMER switch on the overhead switch panel. The bus is tied directly to the battery via a relay and may be activated only when the battery master switch is turned OFF.

The EMERGENCY bus provides power to items in “Chart 1”, and It is used for such things as filing flight plans prior to turning the battery master switch ON or providing emergency power to systems required to land the aircraft in the event of a total electrical failure.

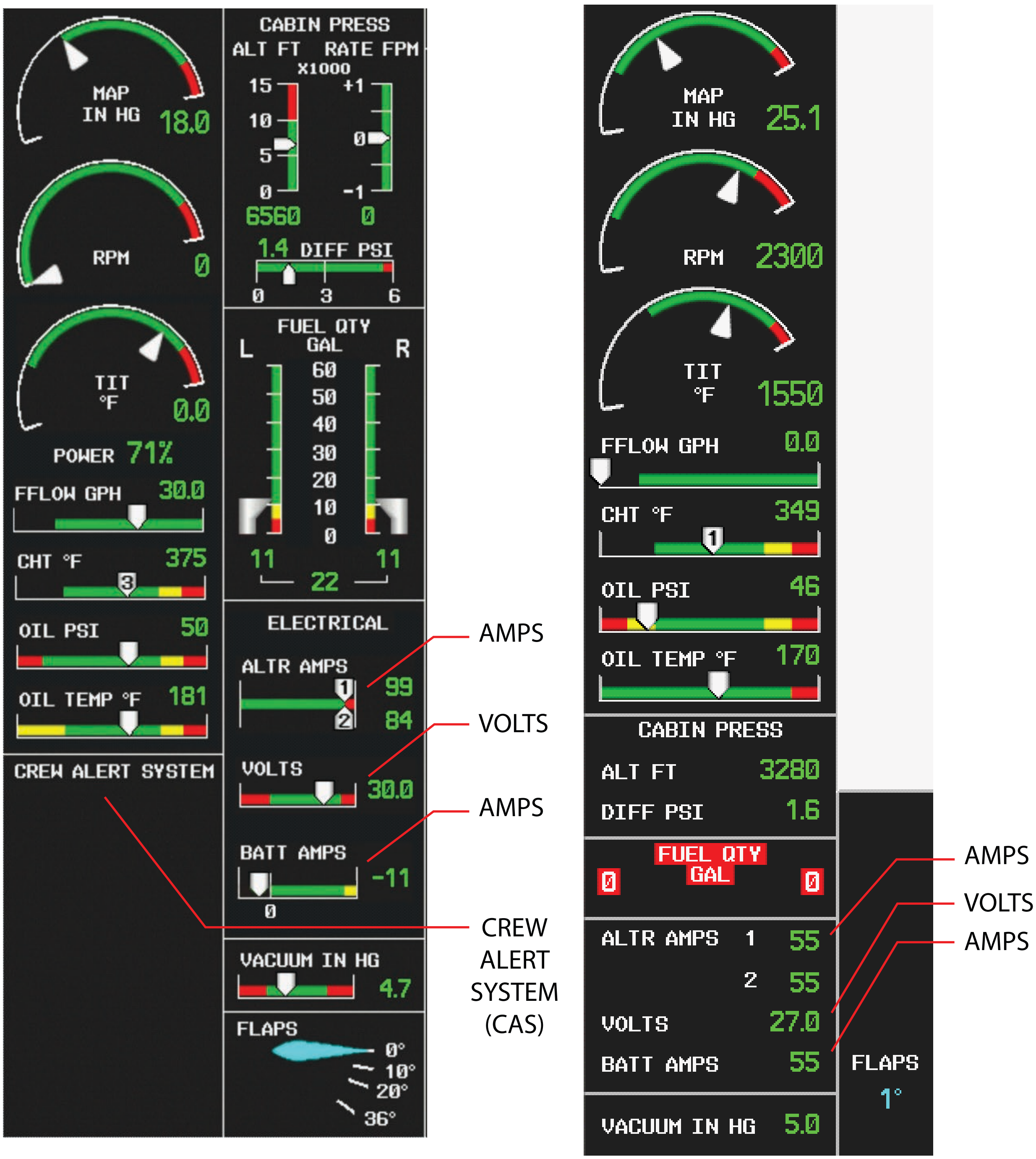
When the Battery Master switch, located on the overhead switch panel, is turned ON, the battery contactor closes, enabling current to flow from the battery to both the start contactor and the tie bus, located on the lower right of the co-pilots instrument panel. Should the airplane’s battery be depleted, a receptacle (located inside the forward baggage compartment door) permits using an external 24 Vdc power source for engine start. With the Battery Master switch OFF, connecting an appropriate external source completes a circuit that closes the external power contactor, permitting current to flow from the external source direct to the starter contactor and the tie bus. Whether using the airplane’s battery, or external power, tie bus overcurrent protection is provided by the 80 ampere tie bus Battery circuit breaker and a 250 amp in-line current limiter fuse.

Alternator amps, battery amps, and voltage are all displayed on the MFD (see “Figure 1”) in Normal Mode or the PFD when the display backup (Reversionary) mode is enabled.

NOTE: When using only the airplane’s battery or a 24 volt external power source, the “VOLTS” CAS message will be displayed on the MFD (Garmin G1000 /G1000 NXi installation). Check the appropriate display for actual voltage.

CHART 1
EMERGENCY BUS SYSTEM (G1000 SHOWN)

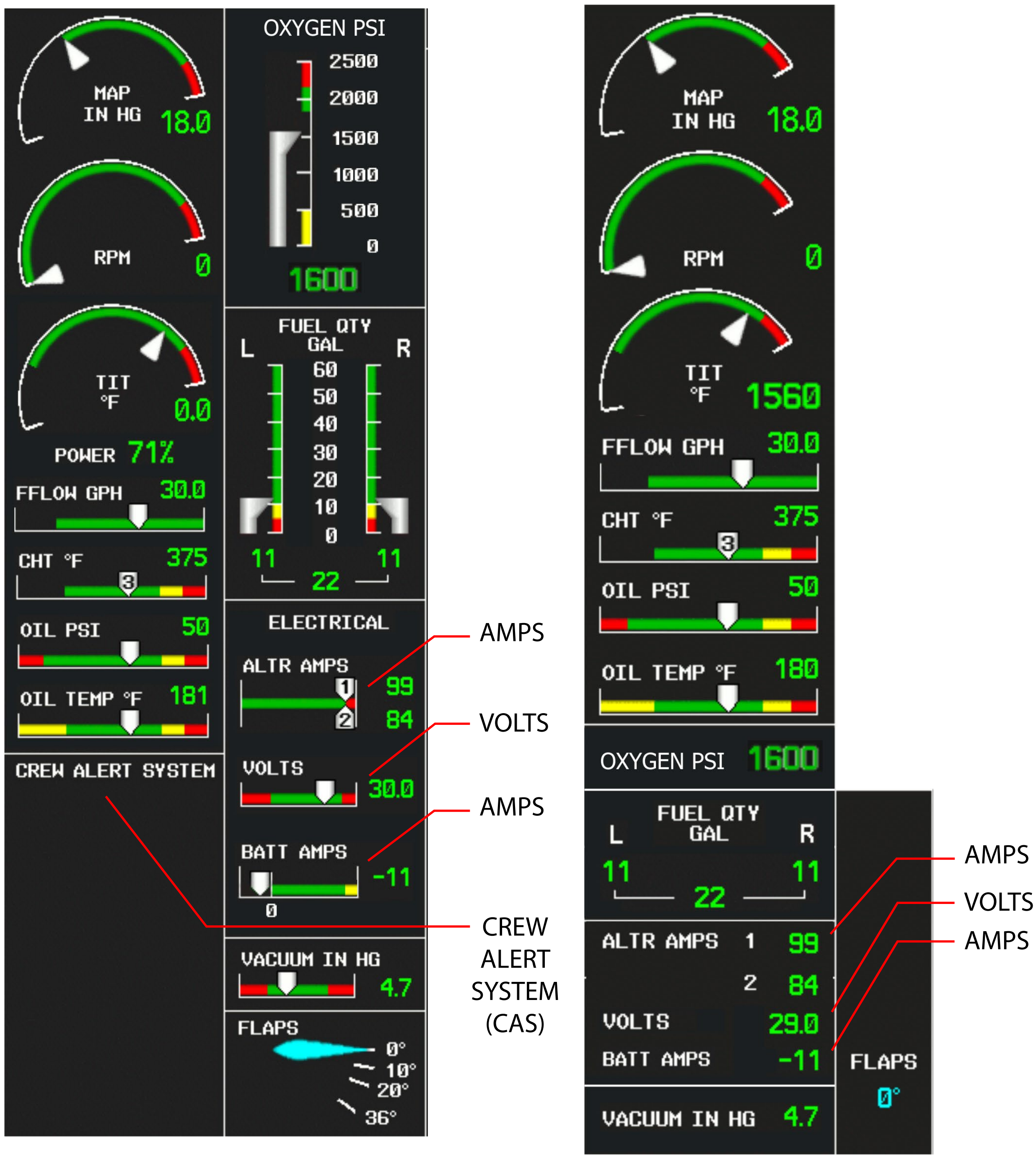
EMERGENCY BUS
GMA
GEA
PFD 1
AHRS 1
ADC 1
INTEG AV1
COMM 1
EMER POWER
GEAR INDICATION
EMER LIGHTS



Normal Mode (MFD)

Reversionary Mode (PFD)

Electrical Displays - Mirage (G1000 shown)
Figure 1



Normal Mode (MFD)

Reversionary Mode (PFD)

Electrical Displays - Matrix (G1000 shown)
Figure 2

2. Contactor Maintenance

The following incorporates Piper Service Letter No. 1116A.

CAUTION: APPLYING EXCESSIVE TORQUE TO THE TERMINALS OF A STARTER CONTACTOR CAN RESULT IN INTERNAL DAMAGE TO THE CONTACTOR. THIS DAMAGE CAN CAUSE THE STARTER TO ENGAGE WHEN THE MAIN POWER BUS IS ENERGIZED. THIS WILL RESULT IN AN UNCOMMANDED ROTATION OF THE PROPELLER, POSING A SERIOUS SAFETY RISK TO ANYONE IN THE PROPELLER'S PATH.

Service spares replacement contactors, whatever device they are installed onto, are configured out-of-the-box as shown in "Figure 3", below. Contactors are labeled with instructions in red text that read:

WARNING: DO NOT REMOVE INNER NUT WHEN ASSEMBLING. USE WRENCH TO HOLD INNER NUT IN PLACE WHEN APPLYING OUTER NUT.

The "inner nut" refers to nuts that are supplied with the contactors. When installing wire terminals onto the threaded posts, two wrenches must be used: One wrench (the back-up wrench) is placed on the inner nut to hold it stationary; the other wrench turns and tightens the outer nut.

Piper models may have contactors manufactured by either White-Rodgers or Lamar. The correct nut torques, by manufacturer, are:

White-Rodgers:

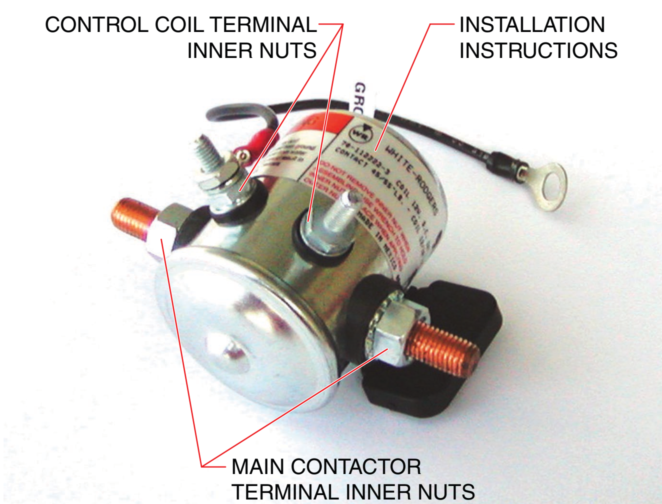
- On the main contactor terminal nuts (5/16 – 24 thread), use 45 to 55 inch-pounds (in-lb).
- On the control coil terminals nuts (#10 – 32 thread), use 12 to 18 in-lb.

Lamar:

- On the main contactor terminal nuts (5/16 – 24 thread), use 35 to 45 in-lb.
- On the control coil terminals nuts (#10 – 32 thread), use 10 to 15 in-lb.

NOTE: The proper installation torque for nuts installed on the threaded terminal posts is also labeled on the contactor.

NOTE: Fastener torques shown above reflect the specifications of the contactor manufacturer at the time of publication of this maintenance alert service letter, and are subject to change. Adhere to vendor installation instructions provided with any Piper approved replacement contactor.



Typical Contactor
Figure 3

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AC GENERATION

1. 110V AC Inverter

A 110 VAC Inverter is located under the forward-facing bench seat. It supplies power to 110 VAC outlets, located at the aircraft centerline. In S/N's 4636001–4636760, two AC outlets are installed: one AC outlet is mounted on the forward side of the main spar cover and a second is mounted to the forward side of the frame at F.S. 218.62. In S/N's 4636761–4636775, a single AC outlet is installed and mounted to the forward side of the frame at F.S. 218.62.

A. Description

NOTE: This inverter can support up to three AC outlet units. Refer to the serial number ranges listed above for the number of used outlets from factory installation.

The In-Seat Power Supply (ISPS) Inverter receives a nominal 28VDC input and converts it to nominal 110VAC, 60Hz output, which is distributed to AC Outlet Units (ACOU) to operate laptop computers and other types of Personal Electronic Devices (PED) that are approved for use in flight. The system is designed to provide up to 225 VA total continuous power, with a maximum of 150 VA to any one ACOU. Remaining available power is automatically apportioned to the remaining outlet if in use. It consists of the following Line Replaceable Units (LRU):

- DC-AC In-Seat Power Supply (ISPS)
- AC Outlet Unit (ACOU)

B. Operation

(1) General

The system provides power as described above. 150VA maximum power per output is internally managed according to power demand and temperature trip points. If the internal temperature limit is exceeded, the ISPS will automatically disable the last user plugged in, until the condition has cleared. This is referred to as “load shedding”. When the temperature has dropped sufficiently, the last user shed will automatically be re-enabled. When the ACOU has power applied, the Status Light is GREEN. If the light is OFF, power is not applied to the outlet.

Ground Fault Interrupt (GFI) circuitry is designed into the ISPS to protect the user from electrical shock. When the GFI circuit is tripped, the ISPS output is disabled and the ACOU Status Light will illuminate WHITE. The GFI condition must be removed and power to the ISPS must be cycled (i.e., cycle the circuit breaker) to reset and initialize the ISPS. (See “Chart 1” on page 24202).

(2) System Power ON

The system powers ON when the master switch is ON and the Inverter circuit breaker (Pilot's Forward CB Panel) is IN. The ISPS is enabled by default. To disable the ISPS pull the Inverter circuit breaker (Pilot's Forward CB Panel) OUT.

Each ACOU has a white backlight that illuminates when 28VDC is applied. Each ACOU has a Status Light on the face plate to indicate 110VAC power availability and ISPS fault status. “Chart 1” on page 24202 lists the different indication conditions.

The ACOU power up indications will differ if the ACOU is powered up with a plug inserted or not. Power up indications are as follows:

(a) ACOU without plug inserted:

1) Status Light will flash 5 times (once per second).

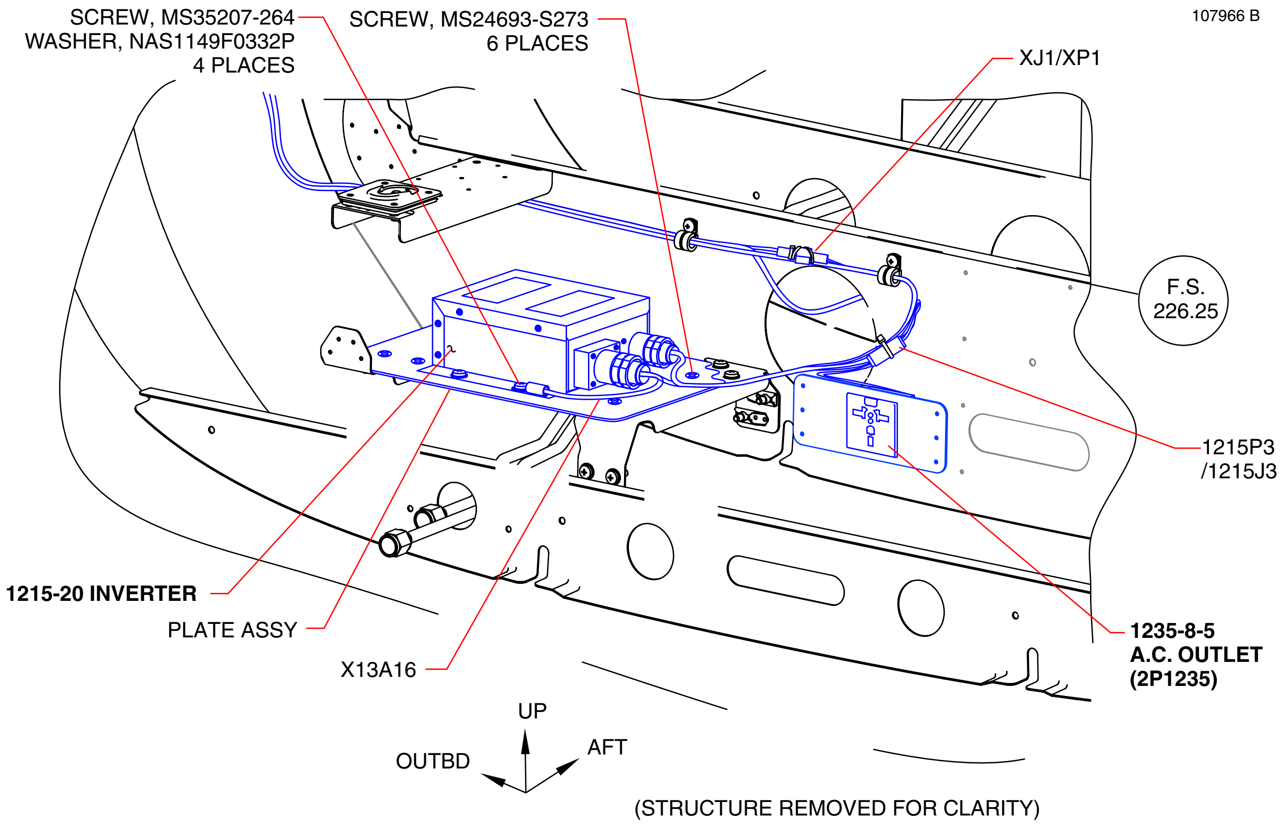
2) Status Light will turn off. This indicates there is not 110VAC at ACOU. If a valid plug insertion is detected, the Status Light will illuminate GREEN and 110VAC will be applied to the ACOU.

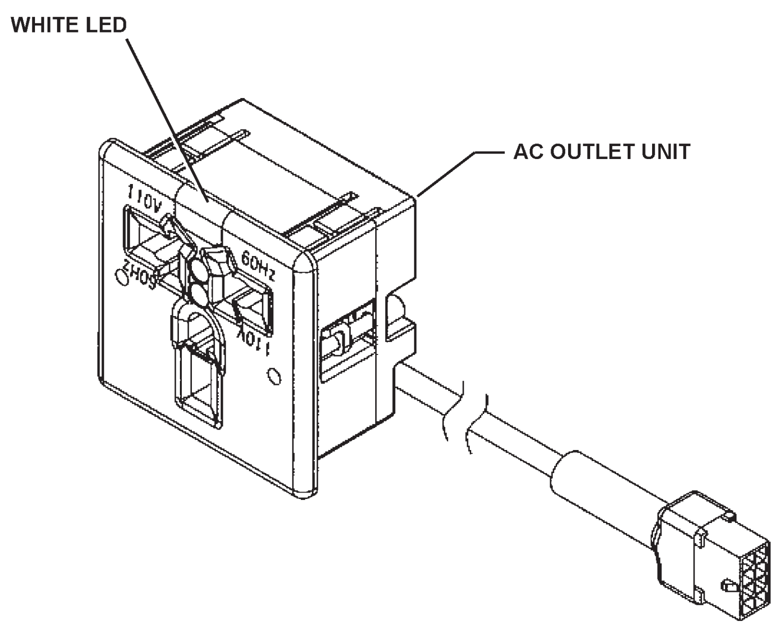
(b) ACOU with plug inserted:

When power is applied the Status Light will flash 5 times, pause, flash 5 more times, and illuminate solid GREEN. This indicates 110VAC is now applied to the ACOU.

CHART 1
AC OUTLET UNIT STATUS LIGHT INDICATIONS

Status Light	Status	Notes
OFF	Power is NOT applied to ACOU. Improper connection of device when user plugs in. ISPS over-current detected, or load shedding.	These faults (except load shedding) are recoverable by removing the plug from the ACOU and re-inserting. Other outlets are not affected.
GREEN	110VAC is applied to the ACOU.	Normal Operating Condition.
FLASHING GREEN (5 seconds)	ISPS is initializing or is functioning properly. Power will be available when a plug is inserted. Auto-reconnect condition: 1) The ISPS will automatically reconnect a user that had been "shed" due to excessive temperature or load, once that condition is no longer present. 2) At turn-on, following the Power On Self Test, if a valid connection insertion already exists.	Normal Power up Sequence. Other ACOUs are not affected.
WHITE	ISPS BIT Failure. GFI Trip. ISPS over-temperature condition.	Faults are recoverable only by cycling input power or reinitializing the ISPS via the Enable/Disable. It is also possible to show WHITE if an ACOU is non-functional (broken wire or stuck plunger) or all ACOUs are effected (no power is available).





110V AC Inverter Installation
Figure 1

Effectivity
4636001–4636775

(3) Operational States

(a) Power-On-Reset

Upon application of AC power the “Power-On-Reset” state is initiated. During this state, the processors are reset. After the Power-On-Reset timer has expired, the processors are released from reset and the ISPS then transitions to the “Initialization” state.

The ISPS operates in one of three states after the “Power-On-Reset” completes.

(b) Initialization

Upon release of the processors from reset, the “Initialization” state begins. During this state, the ISPS initializes hardware and software for operation. Power-on Built-In-Test (BIT) functions are performed.

BIT Functions

a) Verify internal power supply outputs are within specified limits.

b) Determine which OUs are physically connected.

c) Verify the status of PED connections (either both plugs are inserted, no plugs inserted, or only one plug is inserted indicating a fault condition.

d) Verify outlet relays are functional.

e) Verify GFI is functional.

f) Verify integrity of program memory images (checksum algorithm).

The ISPS remains in the “Initialization” state until the input power is greater than 18 volts and the BIT functions are complete. At that time, the ISPS enters the ON state unless any one of the following fault conditions is detected:

Fault Conditions Aborting Initialization

a) A ground fault on any outlet.

b) An over-temperature condition.

c) Failure of a BIT function, including program memory verification, outlet relay function, GFI detection circuit function, and internal power supply voltage regulation.

d) An over or under voltage condition on any output.

The ISPS completes initialization and is available for normal operation within 10 seconds after the Power-On reset time interval has expired.

(c) On State

Upon exit of the “Initialization” state with no fault conditions present, the “On” State is initiated. In the ON state, the ISPS is able to provide AC output power to the outlets, as well as outlet indications.

In-Use Indicator outputs are enabled for a period of approximately 1 minute upon entry into the ON state. The In-Use Indicator outputs are individually enabled whenever the corresponding outlet is powered.

During operation, should any one or more of the following fault conditions be detected, the ISPS automatically transitions to the OFF state.

Fault Conditions Termination ON

a) GFI is detected on any output.

b) An over-temperature condition is detected.

c) An out-of-regulation fault on any of the internal power supply voltages is detected.

(d) Off State

Upon exit of the "Initialization" state with a detected fault condition, or upon detection of certain fault conditions while in the ON state, the ISPS will automatically initiate the OFF state.

Conditions Forcing OFF State

a) GFI is detected on any output.

b) Maximum over temperature limit is exceeded.

c) Controller Watchdog has timed out.

d) Internal power supply (e.g., to power internal circuitry) outputs are outside specified regulation limits.

In the OFF state, the ISPS disables AC power to the outlets and causes the In-Use Indicator LEDs on the Outlet Units to flash (or "blink") at a 1 Hz rate. The In-Use Indicator LEDs flash for a period of approximately 5 minutes, at which time they will be turned OFF if the detected fault persists.

The ISPS sets all OU Status Indicators to WHITE within 5 seconds of detection of failures that caused entry into the OFF state.

The OFF state is entered only as the result of the above fault conditions. The only way to restore the ISPS to the ON state from the OFF state is by:

Removing and reapplying DC input power (i.e., cycle the circuit breaker) after removal of the fault condition.

NOTE: If the fault is not cleared, the ISPS will re-enter the OFF State..

(4) Outlet Unit Connection and Disconnection Processes

(a) Detection of a Connected Load

During the ON State enabled mode, a proper passenger load connection is determined when all of the following exist:

- Both power contacts of the outlet socket are mated within 50 msec.
- An output overload condition is not present.
- A ground fault is not detected.

The ACOU power up indications differ if the ACOU is powered up with a plug inserted or not. Power up indications with a pre-existing plug insertion are as follows:

- The Status Light will flash 5 times (once per second).
- The Status Light will turn off.

Upon detection of a connected load, the ISPS provides AC power to the outlet after a delay of approximately 1.25 seconds (assuming no disconnect is detected during that time) and sets the Outlet Unit status indicator to Green indicating power is applied to the outlet unit.

Upon detection of an incorrect connection, which consists of single pin insertion, dual pin insertion outside the 50msec allowance, or an output overload condition, the ISPS will not connect AC power to the OU. The Status Indicator also remains OFF until a proper connection is made.

NOTE: This is different than the ISPS response at turn-ON, which will initiate a white fault light if an invalid connection is detected.

(b) Detection of a Disconnected Load

A passenger load "disconnection" is declared when either or both power pins of the load connector are unmated for >10msec.

Upon detection of a disconnected load, the ISPS automatically disconnects AC power from the outlet within approximately 110 msec.

(c) Auto-Reconnect Criteria

To prevent a user from having to un-plug and re-plug their device, the ISPS will automatically re-connect existing users under the following conditions:

1) At turn-on, if the ISPS detects that a valid insertion already exists, per the sense contacts in the OU, the ISPS will automatically reconnect.

2) Thermal control – The ISPS will automatically re-connect following software load shedding or the middle thermal trip limit if the ISPS cools sufficiently.

3) After a processor “Hold up” condition has cleared. “Hold up” is defined as an input voltage drop-out of up to 200ms.

Timing: The sampling period to determine the above conditions is 10 ± 1 seconds. The ISPS shall flash the OU status light GREEN for 5 seconds (within the 10 second sample period) at a 1Hz rate prior to applying power to the outlet (to indicate power is available and about to be re-stored to that outlet). Only one user at a time is reconnected per the sample period (10 seconds).

(5) ISPS Protection

Safety features include fused input, overload protection, 115VAC isolation, power switching controlled by the OU and Ground Fault Interrupt to reduce shock hazards. The ISPS provides “latch-off” protection that disables all outputs for the fault conditions listed below.

- GFI fault (any output).
- High Thermal Limit fault (internal to ISPS).
- Watchdog Timeout fault (internal to ISPS).

To recover from these conditions, the operator must remove and reapply DC input power (i.e., cycle the circuit breaker) after removal of the fault condition.

The ISPS provides protection that disables outputs but does not latch-off for the fault conditions listed below.

- Output Over-Voltage fault in the ON state.
- Output Under-Voltage fault in the ON state.
- Short Circuit Protection.
- Over-Load Protection.

Each device connected to each ACOU must be disconnected and properly re-connected to the OU before power is available to the device after this type of fault has been detected.

(a) Over-Load Protection

Each AC output is protected from over-loads with redundant power limiting mechanisms controlled by both software and hardware.

1) Software

The output power is limited to approximately 175 VA per output by the ISPS internal software. If the output power exceeds this limit the software will disconnect power from that outlet. The ISPS software will automatically re-connect if the overload is removed the output voltage will be restored.

If both outputs are active, it can take up to 2.4 seconds to compute the averaged output power for each outlet.

2) Hardware

As a failsafe for the software overload protection the ISPS has a second overload shutdown that is implemented in hardware. If the output power reaches approximately 200VA the hardware overload protection will set the ISPS to the OFF state.

To restore the ISPS to the ON state from the OFF state: remove and reapply DC input power (i.e., cycle the circuit breaker) after removal of the fault condition.

NOTE: If the ISPS has not cooled sufficiently, the ISPS will re-enter the OFF State.

(b) Short Circuit Protection

The ISPS incorporates instantaneous current protection to limit maximum short circuit current to no more than 6 amps peak (combined load for all three OUs, if installed). The AC output may provide up to three full cycles at maximum current (6 amps peak) before shutdown occurs.

(c) AC Output Over/Under Voltage Protection

The 1215 ISPS contains an AC output over/under-voltage protection circuit.

The under-voltage protection function disables AC outputs when an output of less than 99Vrms is detected. The over-voltage protection function disables AC outputs when greater than 121 Vrms. These under and over voltage detection thresholds apply to steady state voltages. The response time of the protection function is approximately one second.

In the event of an over/under voltage condition, all connected outputs are disabled. Each device must be disconnected and properly re-connected to its OU before power is restored to the device.

(d) GFI Protection Circuit

The ISPS provides GFI detection/protection and will latch all power off when a ground fault condition is detected. A ground fault current of 6mA will trip the unit into the off state in 30 msec maximum. The ISPS requires that input power be re-cycled prior to resetting the latched off condition.

(6) Thermal Management

The ISPS performs a comprehensive thermal monitoring and power management function to ensure safe operation of the ISPS. The unit has Low, Middle and High Thermal limits with slightly different protective actions for each.

If the Low Thermal Limit is exceeded the ISPS disconnects each user other than the first user. As the OUs are disconnected, the OU status Indicators are set to OFF in the reverse order of their connection and at an approximate 10 minute disconnect interval.

If users were disconnected due to a Low Thermal Limit and the temperature has subsequently dropped, the ISPS will re-apply power to the outlet unit(s) with valid connections (individually) at approximately 10 second reconnect intervals. The OU status indicators are set to ON as the users are reconnected.

The ISPS disconnects all outputs if the Middle Thermal Limit is exceeded, but continues to operate in the ON state. The ISPS software monitors the Middle Thermal Limit, and performs a reconnect when the ISPS cools sufficiently.

The ISPS disconnects all outputs and causes a transition to the OFF state if the High Thermal Limit is exceeded. The ISPS latches to the OFF state if this condition is detected and will not connect AC power to the OUs until power is reset and the high temperature fault condition no longer persists.

C. Troubleshooting

See “Chart 3” on page 246014.

D. Removal

(1) Inverter

(a) Set the battery/master switch to the OFF position and ensure no external power is connected.

(b) Remove the forward-facing bench seat per Removal under Forward-Facing Passenger Seat in 25-20-00.

(c) Disconnect the electrical connectors from the inverter.

(d) Remove four screws and washers and remove the inverter.

(2) AC Outlet Unit (ACOU)

The AC Outlet Unit is face-mounted. The unit is secured in position by a captured rotating lug which engages the mounting plate from behind after the outlet unit is in position.

(a) Set the battery/master switch to the OFF position and ensure no external power is connected.

(b) Using a 3/32 inch Allen wrench, loosen the two mounting screws enough to allow the rotating lugs to retract.

(c) Pull the unit out of its mounting hole until the electrical harness connecter is accessible.

(d) Disconnect the electrical harness connector being sure to capture/tie-off the pigtail coming out of the spar cover or interior panel.

(e) Remove the ACOU.

E. Installation

(1) Inverter

(a) Position inverter in on mounting plate and secure with four washers and screws.

(b) Connect the electrical connectors to the inverter.

(c) Ensure that all installed outlets have been connected to the inverter before proceeding.

(d) Set the battery/master switch to ON position.

(e) Verify 110 VAC at each of the AC outlets.

1) If 110 VAC present at each of the outlets, proceed to step (f).

2) If 110 VAC not present at each of the outlets, troubleshoot per “C. Troubleshooting”.

(f) Install the forward-facing bench seat per Installation under Forward-Facing Passenger Seat in 25-20-00.

(2) AC Outlet Unit (ACOU)

(a) Connect electrical harness connector.

(b) Feeding the exposed electrical harness into the mounting hole, slide the ACOU into position in the mounting hole.

(c) Using a 3/32 inch Allen wrench, carefully tighten the two mounting screws. Ensure the rotating lugs extend and engage the back of the mounting plate.

(d) Torque to 4–6 in·lbs.

(e) Ensure that the inverter has been installed and connected to the outlet before proceeding.

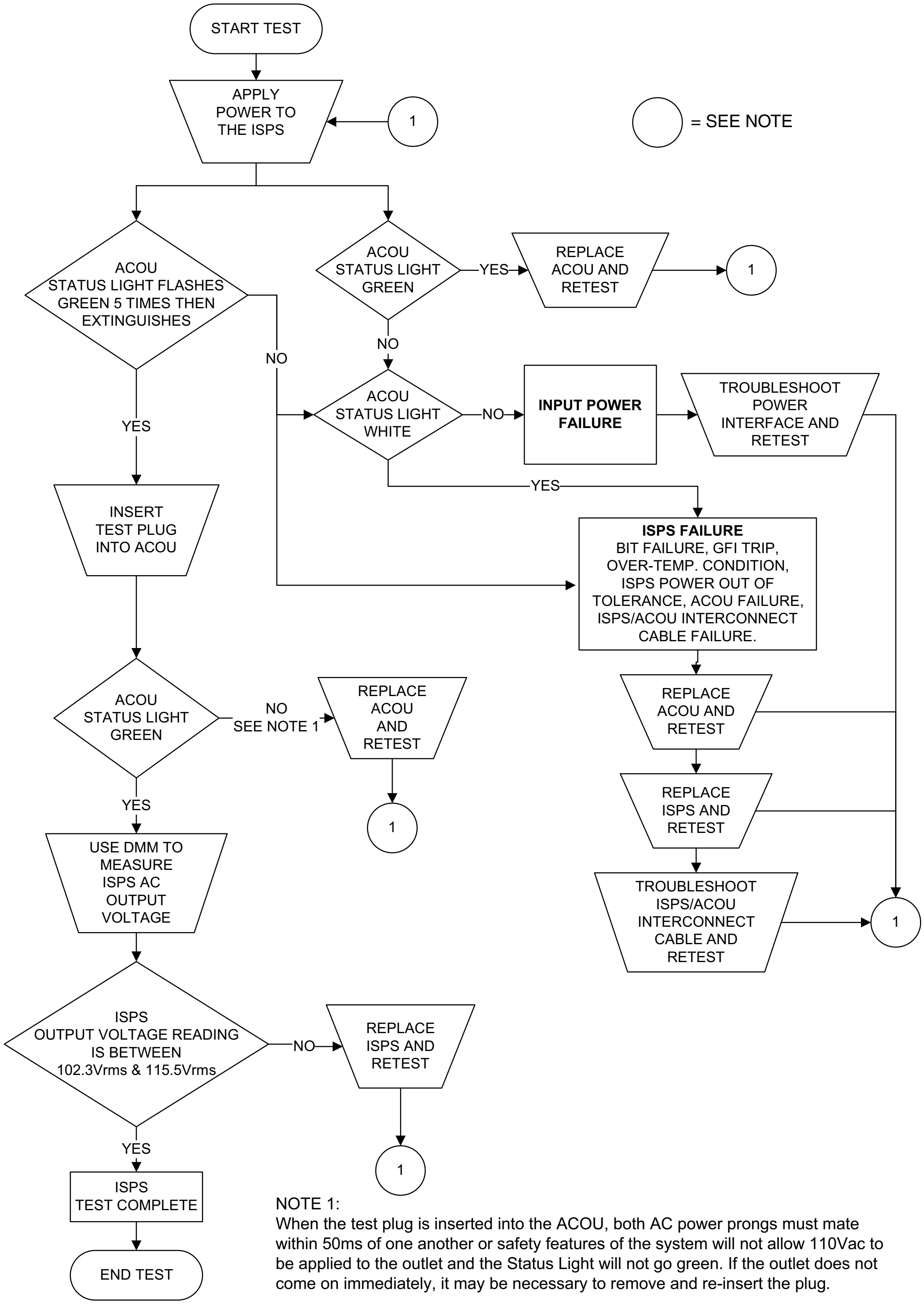
(f) Set the battery/master switch to ON position.

(g) Verify 110 VAC at the AC outlet.

1) If 110 VAC present at the outlet, install is complete.

2) If 110 VAC not present at the outlet, troubleshoot per “C. Troubleshooting”.

CHART 2
TROUBLESHOOTING - 110 VAC INVERTER



NOTE 1:
When the test plug is inserted into the ACOU, both AC power prongs must mate within 50ms of one another or safety features of the system will not allow 110Vac to be applied to the outlet and the Status Light will not go green. If the outlet does not come on immediately, it may be necessary to remove and re-insert the plug.

F. Testing

The following tests verify proper functioning of the ISPS.

(1) Required Equipment

(a) A test cable fabricated per “Figure 2” on page 242011.

(b) A calibrated Digital Multi-Meter (DMM) capable of AC voltage measurement.

(2) Functional Test Procedures

Perform tests (1) and (2) on each ACOU, test (3) is a system test that only needs to be done once per ISPS.

(a) DMM Outlet Voltage Test

1) Apply power to the ISPS.

2) Verify the White Backlight is ON.

3) After the ISPS completes the POST (Power On Self Test): Verify the Status Light flashes 5 times, and then extinguishes.

4) Connect the DMM to the ACOU Test Cable JI and J2.

5) Set the DMM to read AC voltage.

6) Insert the ACOU Test Cable male plug into the ACOU.

NOTE: When the test plug is inserted into the ACOU, both test plug prongs must mate within 50ms of one another or safety features of the system will not allow 110VAC to be applied to the outlet and the Status Light will not illuminate green. If the Status Light remains off, it may be necessary to remove and re-insert the plug.

7) Verify the Status Light illuminates green.

8) Verify the voltage displayed on the LCD is between 102.3VAC and 115.5VAC (110.0Vrms +5%/-7%).

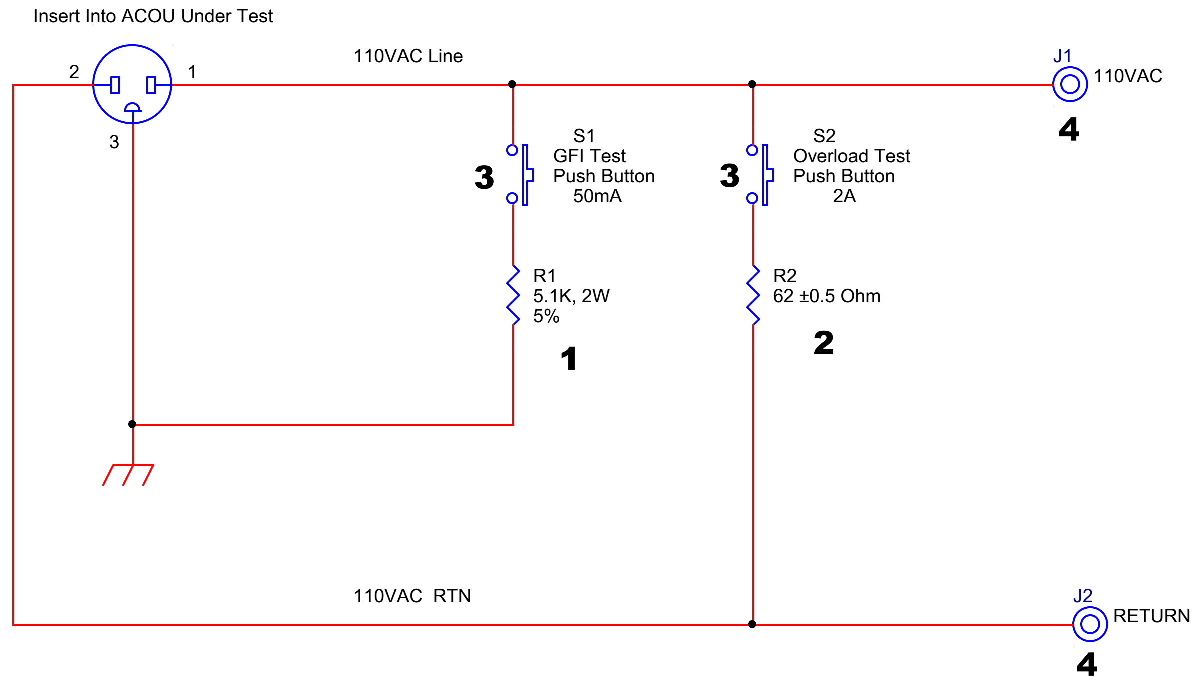
(b) Overload Test

1) Press the Overload Test Switch (S2) and hold down for 2-3 seconds. The DMM should indicate output voltage drops to 0VAC.

NOTE: If the DMM does not drop to zero after three seconds, release switch S2 to limit heating of R2.

2) After S2 is released verify the Status Light flashes 5 times, pauses, flashes 5 more times, and then illuminates.

3) The DMM should indicate output voltage returns to between 102.3VAC and 115.5VAC (110.0Vrms +5%/-7%).



Sample Parts List

Index	Qty	Part Number	Description	Supplier
1	1	5.1KW-2-ND	RES 5.1K OHM 2W 5% METAL OXIDE	Digikey
2	1	AVT200-75-BKTS	RESISTOR POWER ADJ 75 OHM 225W	Huntington Electric
3	2	679-1099-ND	SWITCH PB SPST SEALED IP67 BLACK	Digikey
4	1	461-1215-ND	BANANA JACK PNL MT SET/2 RED/BLK	Digikey
-	1	AC-1414	BOX, ALUM, 14” X 6” X 3”	Bud Industries
-	1	BPA-1514	BOX, BOTTOM COVER	Bud Industries

Fabricated Test Cable
Figure 2

(c) ISPS System Test: GFI

Continuing from the end of step (2):

1) Press the GFI Test Switch (S1).
2) Instantly the DMM should indicate output voltage drops to 0VAC.
3) The Fault light should illuminate: (WHITE)

NOTE: This is a latching fault and can only be cleared by removing the +28VDC and repowering the ISPS (i.e., cycling the circuit breaker).

4) Reset the ISPS by cycling the Enable/Disable switch .
5) After the ISPS completes POST verify the Status Light flashes 5 times, pauses, flashes 5 more times, and then illuminates.
6) Remove the DMM / ACOU Test Cable from the ACOU.
7) Verify the Status Light extinguishes.
8) Remove power to the ISPS system.

(d) End of Test

2. 115V AC Inverter

Optional in S/N's 4636776 and up.

See "Figure 3".

A 115V AC Inverter is located under the forward-facing bench seat. It supplies power to one 115V AC outlet at the aircraft centerline mounted on the forward side of the frame at F.S. 218.62 (at the forward edge of the bench seat).

A. Description

NOTE: This inverter can support up to three AC outlet units. Only one is used in the factory installation.

The 115V AC Inverter receives a nominal 28V DC input and converts it to nominal 115V AC, 60Hz output, which is distributed to a single AC Outlet to power a variety of Personal Electronic Devices (PED), such as a laptop computer, that are approved for use in flight. The system is designed to provide 250VA of continuous power. It includes short circuit protection, overload capability, low voltage shut-down, temperature monitoring, and fault indication to the Engine/Airframe Unit (GEA).

No troubleshooting, inspection, maintenance, or calibration is necessary for continued airworthiness of the inverter. If the unit fails to perform to specifications, the unit must be replaced or serviced by Mid-Continent Instruments and Avionics or their authorized designee.

The inverter system consists of the following Line Replaceable Units (LRU):

- Inverter
- AC Outlet

B. Operation

(1) General

The system provides power as described above. The system includes an overload capability up to 375VA for over 30 minutes until the internal temperature limit is exceeded. If the internal temperature limit is exceeded, the DC-AC Inverter will automatically disable the output and the fault output will indicate. The inverter will continue to remain shut-down in a fault condition until the temperature returns to within acceptable limits. This over-temperature reset occurs automatically without external intervention required. Ground Fault Protection (GFP) circuitry is designed into the DC-AC Inverter to protect the user from electrical shock. When the ground fault is tripped, the AC output will be disabled and the fault output will indicate.

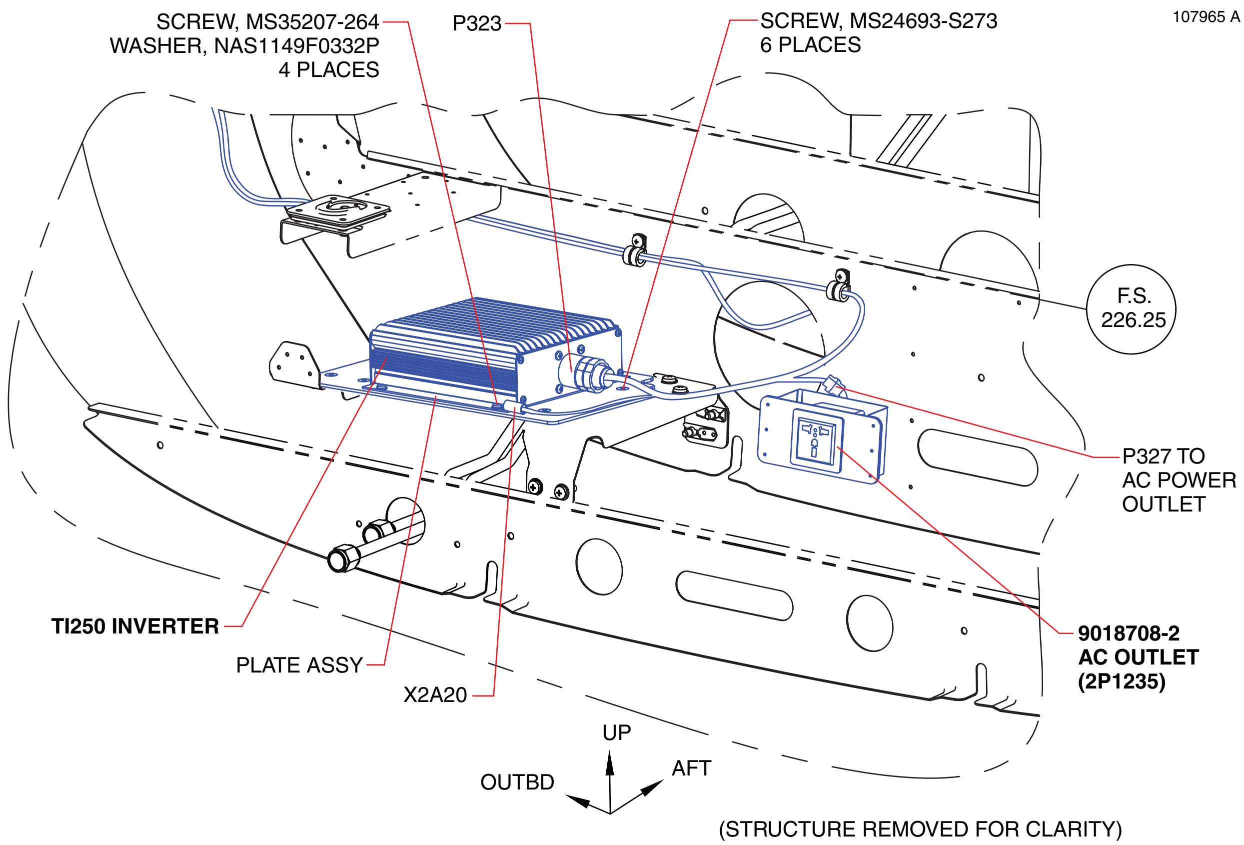
(2) DC-AC Inverter Protections

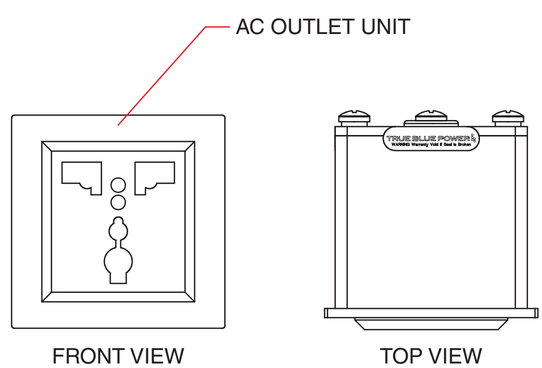
(a) Overvoltage Protection

When the input voltage exceeds the operating range of the unit (36V DC) the unit senses an over-voltage condition and disables the output. The unit will dynamically monitor the input voltage such that if the input returns to within the normal operating range, the output will be enabled and allow the unit to operate normally. Over-voltage will not activate the fault status output.

(b) Under-Voltage Protection

When the input voltage drops below the operating range of the unit (20V DC) the unit senses an under-voltage condition and disables the output. The unit will dynamically monitor the input voltage such that if the input returns to within the normal operating range, the output will be enabled and allow the unit to operate normally. Under-voltage will not activate the fault status output.





Effectivity
4636776 & up

115V AC Inverter Installation
Figure 3

(c) Over-Temperature Protection

The inverter contains an internal temperature sensing device that continually provides monitoring and feedback to the control circuits. When the unit senses an internal condition that exceeds maximum temperature ratings, the output is disabled and the fault output will indicate. The inverter will continue to remain shut-down in a fault condition until the temperature returns to within acceptable limits. This over-temperature reset occurs automatically without external intervention required.

(d) Short-Circuit and Over-Current Protections

The inverter is capable of surviving a short circuit or over-current event without permanent damage or effect to long-term reliability. The unit can provide over its rated power output up to 375 watts for over 30 minutes (until over-temperature protection shutdown occurs). The inverter monitors the AC output to determine a short-circuit or over-current situation. If detected, the power output is limited. If the short-circuit or over-current situation continues for more than a few seconds, the inverter will turn-off the AC output and indicate a fault on the fault output.

(e) GFP Protection

The inverter monitors the AC output for indications of current imbalance between the two AC output lines. Should a ground fault be detected, the AC output will be disabled and the fault output will indicate. Remove all inverter loads before re-enabling the inverter.

(f) Fault Shutdown and Indication

The inverter monitors the AC output load, its internal temperature, and operation status. Should the load be excessive or the inverter become too hot, the inverter will disable the AC output and indicate a fault status on the fault output.

To reset the inverter and clear the fault:

1) Remove all inverter loads before re-enabling the inverter.

2) Re-enable the inverter by switching the inverter OFF for four seconds, and then back to ON.

NOTE: If the inverter is over-temperature it will not re-enable until it has cooled. Fault can also be cleared by cycling power to the inverter.

C. Removal

(1) Inverter

(a) Set the battery/master switch to the OFF position and ensure no external power is connected.

(b) Remove the forward-facing bench seat per Removal under Forward-Facing Passenger Seat in 25-20-00.

(c) Disconnect the electrical connector from the inverter.

(d) Remove four screws and washers and remove the inverter.

(2) AC Outlet

The AC Outlet is face-mounted. The unit is secured in position by a captured rotating lug which engages the mounting plate from behind after the outlet unit is in position.

(a) Set the battery/master switch to the OFF position and ensure no external power is connected.

(b) Using a 3/32 inch Allen wrench, loosen the two mounting screws enough to allow the rotating lugs to retract.

(c) Pull the unit out of its mounting hole until the electrical harness connecter is accessible.

(d) Disconnect the electrical harness connector being sure to capture/tie-off the pigtail coming out of the spar cover or interior panel.

(e) Remove the AC Outlet.

D. Installation

(1) Inverter

(a) Position the inverter in on mounting plate and secure with four washers and screws.

(b) Connect the electrical connector to the inverter.

(c) Ensure that all installed outlets have been connected to the inverter before proceeding.

(d) Set the battery/master switch to ON position.

(e) Verify 115 VAC at the AC outlet.

(f) Install the forward-facing bench seat per Installation under Forward-Facing Passenger Seat in 25-20-00.

(2) AC Outlet

(a) Connect electrical harness connector.

(b) Feeding the exposed electrical harness into the mounting hole, slide the ACOU into position in the mounting hole.

(c) Using a 3/32 inch Allen wrench, carefully tighten the two mounting screws. Ensure the rotating lugs extend and engage the back of the mounting plate.

(d) Torque to 4–6 in·lbs.

(e) Ensure that the inverter has been installed and connected to the outlet before proceeding.

(f) Set the battery/master switch to ON position.

(g) Verify 115 VAC at the AC outlet.

DC GENERATION

3. Troubleshooting

See "Chart 1" on page 24302.

A simple in-aircraft check can be made to determine if an alternator warning light illumination is caused by either a low voltage or an overvoltage condition. Turn both alternator switches "OFF" and then back "ON" again, one at a time. If the alternator annunciator(s) extinguish(es), the system(s) can be considered normal and can be verified so by the ammeter indication. In any case, either alternator system output can be read on the appropriate ammeter.

NOTE: Illumination of the alternator warning annunciator at low engine idle speeds, with high electrical loads on the systems, may occur and is considered normal, especially if the battery is requiring a heavy charge. Under such conditions, the annunciator will extinguish at an increased rpm, as the battery charge returns to normal and the electrical load is reduced.

NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.

4. Distribution

See "Figure 1" on page 24308.

CHART 1 (Sheet 1 of 5)
TROUBLESHOOTING (ELECTRICAL SYSTEM)

Trouble	Cause	Remedy
Zero output indicated on ammeter regardless of rpm (refer to alternator system test procedure).	Open field circuit.	With the battery switch turned on, check for battery voltage from the airplane's main bus through the entire field circuit to the alternator field terminal. Measure the voltage from the ground (-) to the following points (+) in sequence: bus bar, field circuit breaker (5A), field terminals of master switch voltage regulator and alternator field terminal. Interruption of voltage through any of these points isolates the faulty component or wire which must be replaced. (See wiring schematic, Chapter 91.)
	Open output circuit.	With the battery switch turned on, check for battery voltage from the airplane's main bus through the entire output circuit to the alternator battery post. Measure voltage from ground (-) to the following points (+) in sequence: bus bar, output current limiter, ammeter, and alternator battery post. Interruption of voltage through any of these points isolates the faulty component or wire which must be replaced. (See wiring schematic, Chapter 91.)

NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.

CHART 1 (Sheet 2 of 5)
TROUBLESHOOTING (ELECTRICAL SYSTEM)

Trouble	Cause	Remedy
Zero output indicated on ammeter regardless of rpm. (continued)	Open field winding in alternator.	Disconnect field terminal of alternator from field wiring and check for continuity from field terminal to ground with ohmmeter (20-100 ohms) depending on brush contact resistance.
CAUTION: TURN MAGNETO SWITCH TO OFF BEFORE TURNING PROP.		
		(Pull propeller slowly by hand turning alternator rotor through 360 of travel.)
		If resistance is high, check brushes for spring tension and excessive wear and replace if necessary. If brushes are okay and field reads open, replace alternator.
	Shunt load resistor 5 amp fuse open.	Replace fuse after continuity check to ground.
Output indicated on ammeter does not meet minimum values specified in alternator control unit test procedure.	Faulty voltage regulator.	Start engine, turn on load (ref. alternator test procedure), set throttle at 2300 RPM. Check voltage at buss bar [convenient check point, remove cigar lighter and check from center contact (+) to ground (-). Voltage should be 28 volts minimum. If voltage is below this value, replace regulator.
NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.		

CHART 1 (Sheet 3 of 5)
TROUBLESHOOTING (ELECTRICAL SYSTEM)

Trouble	Cause	Remedy
Output indicated on ammeter does not meet minimum values specified in alternator control unit test procedure. (continued)	High resistance connections in field or output circuit.	Check visually for loose binding posts at the various junction points in system, alternator battery post, lugs on ammeter, connections at voltage regulator, circuit breaker, etc. (See wiring schematic, Chapter 91.)
		Examine crimped terminal ends for signs of deterioration at crimp or strands of broken wire at crimp. Tighten any loose binding posts or replace bad wire terminals.
	Open rectifier.	If any of the rectifiers of the alternator open up internally, it will result in a definite limitation on the current that can be drawn from the alternator. After having checked the previous causes of low output it can be assumed that a faulty rectifier exists.
Field circuit breaker trips.	Short circuit in field circuit.	Disconnect field wiring at terminal of alternator. Turn on master switch. If breaker continues to trip, proceed to disconnect each leg of field circuit, working from the alternator towards the circuit breaker until breaker can be reset and will hold. Replace component or wire which was isolated as defective. (See wiring schematic, Chapter 91.)

NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.

CHART 1 (Sheet 4 of 5)
TROUBLESHOOTING (ELECTRICAL SYSTEM)

Trouble	Cause	Remedy
Field circuit breaker trips. (continued)	Short circuit in field winding of alternator	Disconnect field wiring at terminal of alternator. Turn on master switch. Reset breaker, and if breaker fails to retrip, this isolates short circuit to field of alternator itself. Check brush holders for shorting against frame. If there are no obvious signs of a physical short circuit at field terminal or brush holder, replace alternator.
NOTE: Intermittent short circuit.		
		Internal short circuiting of the field can occur at various positions of the rotor, therefore, reconnect field, reset breaker.
CAUTION: TURN MAGNETO SWITCH TO OFF BEFORE TURNING PROP.		
		Pull propeller slowly by hand turning alternator rotor through 360° of travel. Observe circuit breaker for signs of tripping.
Excessive ammeter fluctuation.	Excessive resistance in field circuit.	Check all connections and wire terminals in field circuit for deterioration such as loose binding posts, broken wire strands at terminals, etc. Tighten all connections and replace faulty terminals.
NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.		

CHART 1 (Sheet 5 of 5)
TROUBLESHOOTING (ELECTRICAL SYSTEM)

Trouble	Cause	Remedy
Excessive ammeter fluctuation. (continued)	High field circuit resistance.	If problem persists, jump across the terminals of the following components one at a time until the faulty unit is isolated. a. Field 5-amp (alternator circuit protector). b. Alternator switch. c. Overvoltage relay.
	Defective voltage regulator.	Replace voltage regulator.
	Faulty grounds.	Completely clean all corrosion from grounding points.

NOTE: The Voltmeter and Ammeter are displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.

5. Alternator System

Refer to "Figure 1" on page 24308.

These airplanes use a dual 70-amp (S/N's 4636001 thru 4636131) or 75-amp (S/N's 4636132 and up, S/N's 4692001 and up) alternator system, consisting of two belt-driven alternators that feed their individual positive outputs to the main bus through separate 60-amp (S/N's 4636001 thru 4636131) or 80-amp (S/N's 4636132 and up, S/N's 4692001 and up) Main Bus circuit breakers. Shunts in each alternator output feed total current flow to the respective ammeter. Each ammeter circuit is protected by two 5-amp fuses.

NOTE: The Ammeters are displayed on the MFD when either Avidyne Entegra or Garmin G1000/ G1000 NXi is installed.

6. Main Electrical Bus

In S/N's 4636001 and up, and S/N's 4692134 and up, the main electrical bus with associated circuit breakers is located on the pilots forward and aft side panels. The non-essential bus with associated circuit breakers is located on the pilots aft side panels. On S/N's 4692001 thru 4692133, the main bus and the non-essential bus with associated circuit breakers is located on the lower right side of the instrument panel.

Current is fed to the main electrical bus by two conductors. Two in-line diodes provide isolation in the event of a ground fault in one of the feeder lines. The two feeders are protected by two 80 amp circuit breakers. The non-essential bus is fed by the 70 amp circuit breaker.

Operational check of bus isolation diodes

A. Turn on external or battery power.

B. Check to see that landing gear indication circuit breaker is in.

C. Check to see that gear position lights are on.

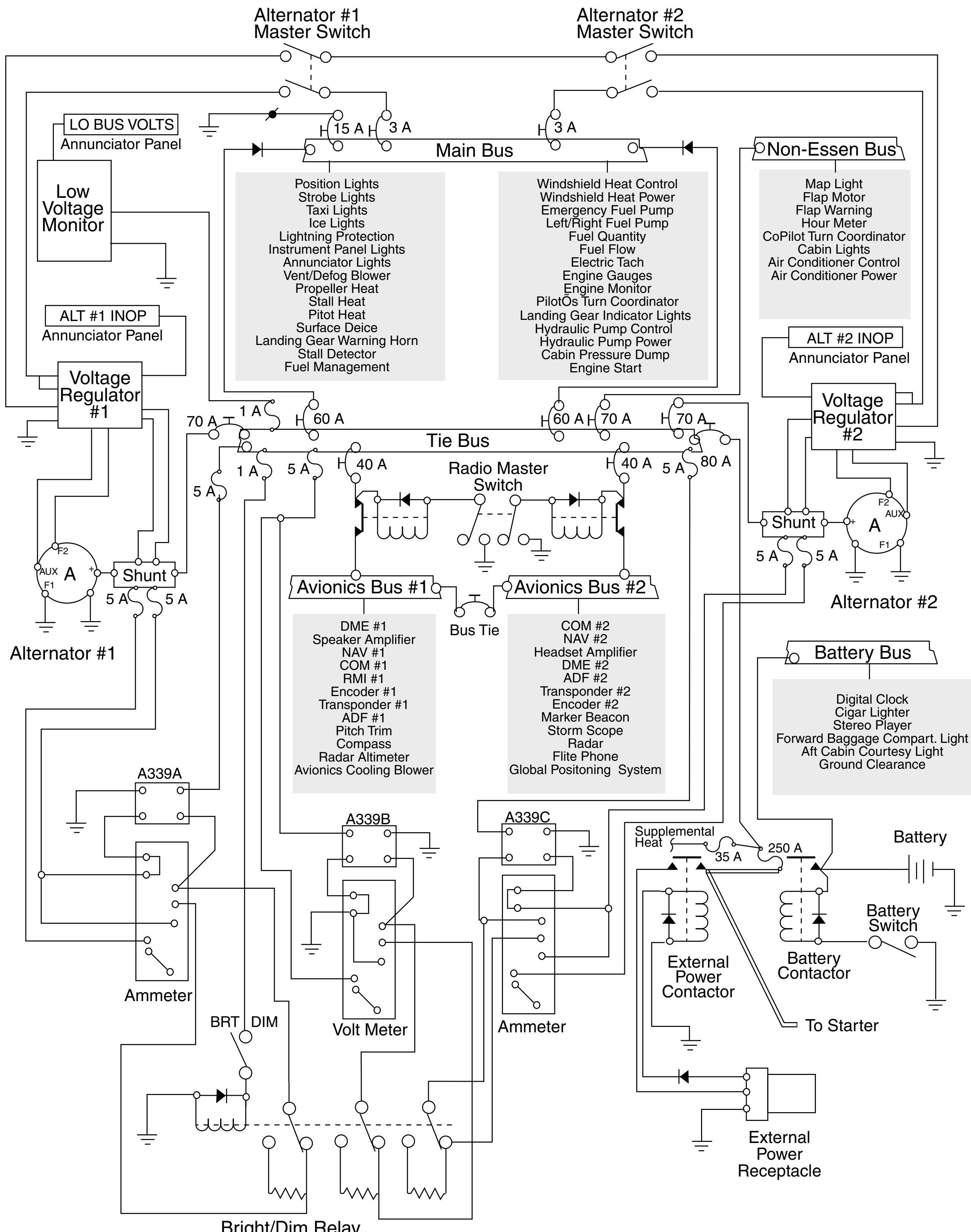
D. Pull one of the Main circuit breakers. The gear position lights should remain on.

E. Reset the Main circuit breaker that was pulled in Step D. Pull the other Main circuit breaker. The gear position lights should remain on.

F. If gear position lights are inoperative when either Main circuit breaker is pulled, troubleshoot the Main electrical bus system.

7. Avionics Bus

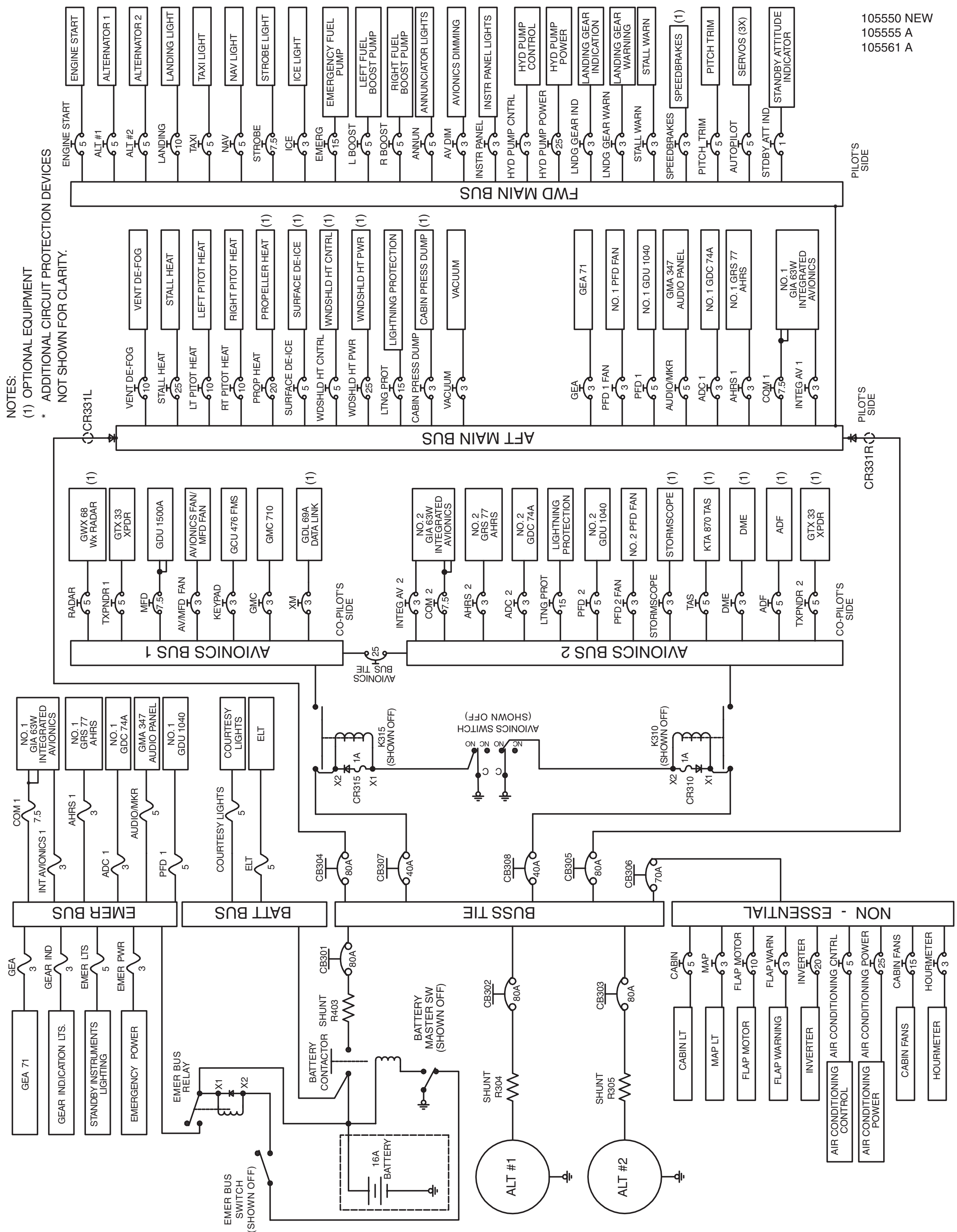
In S/N's 4636001 and up, and S/N's 4692134 and up, separate avionics busses with associated circuit breakers are located on the co-pilots side panel. On S/N's 4692001 thru 4692133, separate avionics busses with associated circuit breakers are located on the lower right side of the instrument panel. These busses are fed through independent contactors. The feeders to the contactors are protected by 40-amp circuit breakers. The busses are interconnected by a 25-amp bus tie and controlled by the avionics/radio master switch. When the AV BUS MASTR switch in the overhead switch panel is pressed, both avionics contactors (normally closed) relax allowing current to flow to both avionics busses. Should the need arise, either avionics bus can be isolated by pulling the avionics bus BUS TIE circuit breaker and the appropriate avionics circuit breaker.



Effectivity
with conventional gauges
or Avidyne Installation

DC Power Distribution, Dual Alternator (Typical)
Figure 1 (Sheet 1 of 2)

24308



DC Power Distribution, Dual Alternator (Typical)
Figure 1 (Sheet 2 of 2)

Effectivity
with G1000 Installation

8. Voltage Regulation and Control

WARNING: USE ONLY AN INSULATED SCREWDRIVER TO MAKE THIS ADJUSTMENT.

Voltage regulation and control are provided for both alternators by Lamar solid-state alternator control units that regulate voltage through control of field voltage and automatic overvoltage protection. The only adjustment is the voltage adjust potentiometer located under the removable cover plug.

System voltage should be set at 28.5 Vdc maximum. Should an overvoltage condition occur, the unit can be reset by cycling the alternator switch "OFF" and back to "ON". The Lamar alternator low output sensor will activate the alternator output warning lamp on the annunciator panel. If an overvoltage trip occurs, an alternator is switched off or an abnormally low voltage exists due to a faulty system.

9. Alternators

Electrical power for the 28 Vdc, negative-ground, dual-fed split-bus electrical system is supplied by two belt-driven, parallel-connected 28Vdc: 70-amp Prestolite alternators, No. ALV-8421-LS (S/N's 4636001 thru 4636131); or, 75-amp Electrosystems alternators, No. ES4039-LS (S/N's 4636132 and up) (S/N's 4692001 and up).

Voltage regulation is provided by a solid-state voltage regulator, which automatically monitors and controls alternator output by controlling the alternator field winding current. Also, an overvoltage control is provided to shut down alternator output should an overvoltage condition occur. The alternator's output can also be shut off manually by individual alternator ON-OFF switches, which interrupt field winding voltage to the voltage regulator.

CAUTION: DO NOT MANUALLY OPERATE (PULL) THE ALTR1 OR ALTR2 CIRCUIT BREAKER WHEN THE ENGINE IS RUNNING AND THE ALTERNATOR IS OPERATING PROPERLY.

Over-current protection is provided by the 70-amp (S/N's 4636001 thru 4636131) or, 80-amp (S/N's 4636132 and up) (S/N's 4692001 and up) tie bus ALTR 1 and ALTR 2 circuit breakers. Two ammeters are fed from taps on each shunt resistor and indicate the individual electrical load of each alternator. Ammeters are located on the lower center section of the instrument panel (S/N's 4636001 thru 4636274); or are included in the MFD on Avidyne Entegra or G1000/G1000 NXi aircraft. Should an overvoltage condition occur in either alternator, its voltage regulator will shut off the field-winding voltage of that alternator; thus overvoltage relays are not required. When either alternator fails, or is selected OFF, the appropriate ALTERNATOR INOP annunciator light will illuminate; or on MFD equipped aircraft, the appropriate ALTERNATOR FAIL Crew Alerting System message will illuminate.

A. Alternator Belt Tension

(PIR-89700, Rev. BK.)

See also Lycoming Service Instruction No. 1129 latest revision.

NOTE: For adjustment of the A/C compressor belt tension, see 21-50-00.

If properly installed, tensioned, and checked periodically, the alternator drive belt will give very satisfactory service. However, an improperly tensioned belt will wear rapidly and may slip and reduce alternator output. Consequently, a belt should be checked for proper tension at the time it is installed, again after 25 hours operation and each 100 hours thereafter, or whenever any work is performed that would affect belt tension.

(1) Span Tension

Adjust the alternator belt tension as follows:

(a) Alternator belt to be installed with span tension of 90-100 lbs.

(b) Run engine for a 15 minute period at 1200 RPM

(c) Shut down engine and recheck. Reset tension to 70 lbs. (if tension falls to 50 lbs.).

(d) This tension check should be made at every 100 hours or annual inspection, whichever occurs first.

(2) Torque Method

This method of checking belt tension consists of measuring torque required to slip the belt at the small pulley as follows:

(a) Apply a torque indicating wrench to the nut attaching pulley to alternator and turn in a clockwise direction. Observe torque shown on wrench at the instant the pulley slips.

(b) Adjust belt to proper tension as specified in “Chart 2”.

(3) Deflection Method

Check belt tension by measuring the amount of deflection caused by a predetermined amount of tension as follows:

(a) Attach the hook of a small spring-scale to the belt at the mid-point between rear gear support and alternator.

(b) Pull scale until it reads 14 pounds. (10 pounds for used belts.)

(c) Measure the distance the belt moved with the 10 or 14 pound load. The distance (deflection) must be 5/16 inch. If less than 5/16 inch, belt is too tight.

(4) Belt Tension Gauge

Use a belt tension gauge such as the Burroughs deluxe model or equivalent.

(a) Attach gauge over the belt per gauge manufacturer’s instructions.

(b) Quickly release handle and read tension.

(c) Repeat steps (1) and (2) several times to eliminate the possibility of an inaccurate reading.

NOTE: Slight variations in readings taken at different locations on the belt are normal.

(d) If a new belt is installed, set tension 25 percent above operating range to allow for stretch that will occur as soon as belt is operated.

(5) Troubleshooting

See “Chart 3” on page 243012 to troubleshoot alternator belt issues.

CHART 2
ALTERNATOR BELT TENSION

Width of Belt	Condition	Torque (Ft.-Lbs.)*
3/8 inch	New	11 to 13
	Used	7 to 9
		* indicated at alternator pulley

NOTE: Higher tension specified for a new belt will compensate for the initial stretch at the first operation. Higher tension values must not be applied to previously used belts.

CHART 3 (Sheet 1 of 2)
TROUBLESHOOTING ALTERNATOR BELTS

Trouble	Cause	Remedy
Belt slippage.	Lack of tension.	Increase tension.
	Overloaded drive.	Reduce load or check pulley size.
	Worn pulley or V grooves.	Replace pulley.
	Oily drive conditions	Clean drive; resolve oil leak.
Belt turns over.	Misaligned drive.	Realign shafts and pulleys
	Worn pulley or V grooves.	Replace pulley
	Heavy impluse loads.	Check idler pulley (spring or not) Check tension
	Excessive vibration.	Tension v-belt. If tension is good, inspect belt condition or replace.
	Broken cords caused by prying belt over pulley.	Replace belts. Do not pry belt over pulley.
	Wrong pulley or belt.	
Rapid belt wear.	Worn pulley grooves.	Replace pulley.
	Pulleys misaligned.	Align the pulleys & brackets.
	Mismatched belts. (Multi belt installations only.)	Replace with matched belts.
	Belt slippage.	Increase tension.
	Pulley diameter too small.	Check P/N and/or replace pulley.
Belt separated.	Foreign materials in drive belt.	Check that pulleys are protected.
	Belt slippage. (over much time.)	Increase tension.
	Heavy start up loads.	Reduce loads before starting.
	Belts damaged during installation.	Install new belts properly.
Belt Stretch.	Excessive drive tension.	Use proper tension.
	Broken cords using multiple belts.	Replace belts with matched set.
	Large misalignment.	Realign brackets and pulleys.
Belt squeal.	Belt slippage.	Increase tension.
	Insufficient arc of contact.	Increase center distance.
	Over loaded drive.	Check pulley size.

CHART 3 (Sheet 2 of 2)
TROUBLESHOOTING ALTERNATOR BELTS

Trouble	Cause	Remedy
Belt chirp.	Misaligned drive.	Align the pulleys & brackets.
	Belt tilted in pulley groove.	Correct angle of belt.
	Belt riding on side of pulley groove.	Adjust bracket in or out to seat belt.
Belt bottom cracks.	High surrounding temperature.	Provide ventilation.
	Pulley diameter too small.	Redesign drive.
	Belt slippage.	Increase tension.
Overheated bearings.	Continous belt slippage.	Increase tension.
	Excessive drive tension.	Tension drive properly.
	Bearings not lubed or damaged.	Replace bearing.

B. Alternator Control Units (Voltage Regulators)

The system uses two Lamar B-00382-1 alternator control units (voltage regulators).

(1) Adjusting Voltage Regulator Units

The only adjustment necessary to maintain the alternator system is the adjustment of the voltage control on the voltage control unit. A voltage of 28.8 Vdc is automatically maintained. All other adjustments are made at the time of installation and need not be reset.

(2) Bench Tests (Lamar B-00382-1)

CAUTION: IN-AIRCRAFT TESTING WITH ALTERNATOR RUNNING IS NOT RECOMMENDED. DO NOT BYPASS REGULATOR BUS TO FIELD AS A MEANS OF CHECKING ALTERNATOR OR OVER-VOLTAGE PROTECTION WITH ALTERNATOR RUNNING.

(a) Set-up

3) Remove unit from aircraft.

4) Set up the following equipment as shown in “Figure 2” on page 243016:

Power:	Pure dc regulated power supply (A) 28.8V 5A
	Adjustable 6-12V power supply (B) 0.05A
Resistors:	R1 1
	R2 650 fine adjustment
	R3 7.2 dummy load 4A
Ammeter:	0-5A dc
Voltmeters:	Precision meter between pins 1 and 8;
	between pin 3 and power (B) ground;
	50 MV precision meter between shunts at pins 6 and 7.
Switch:	Between R1 and R2
Indicator:	28V light (Mazda #1829 or equivalent)

5) Set up load resistance as follows:

a) Jumper pins 1 and 2.

b) Set dummy load resistance for 4A 28.8V (approximately 7.2).

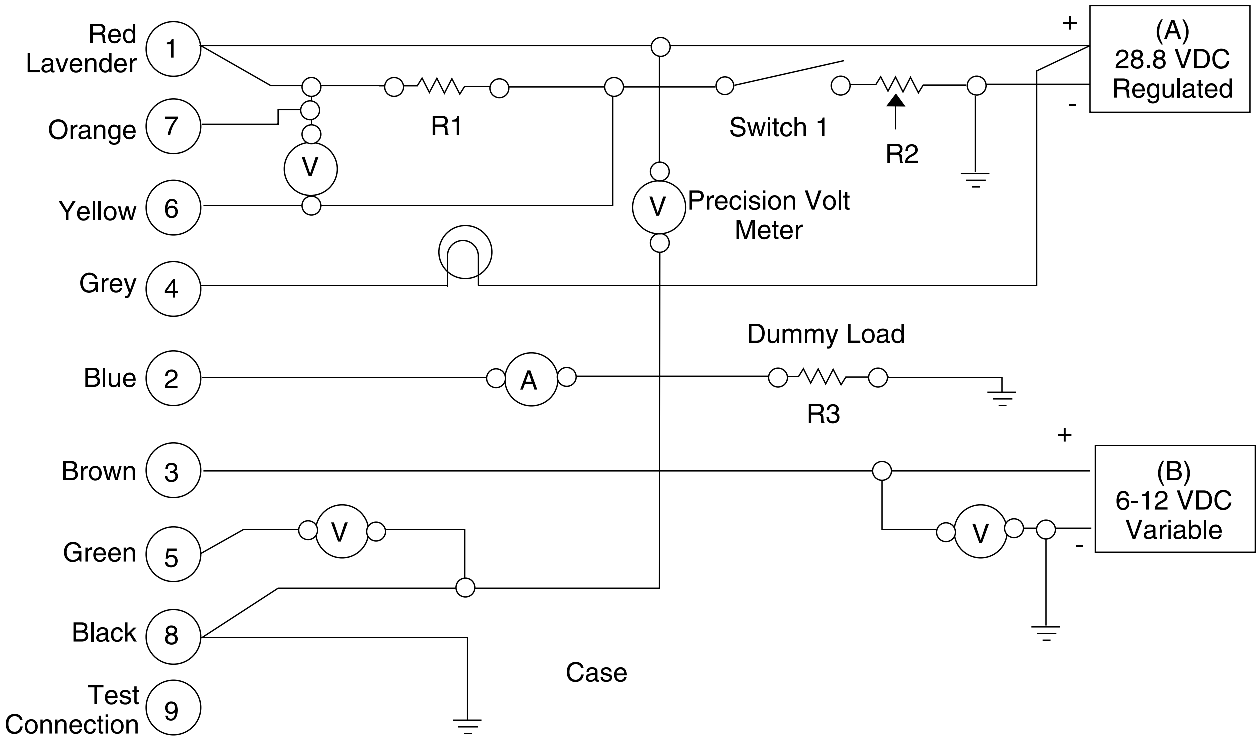
c) Remove jumper.

(b) Voltage Regulator Test

1) Turn power off.
2) Mate unit with test connector.
3) Open switch #1.
4) Apply regulated power supply (A). Hold constant @ 28.8 V ± 0.025 V.

NOTE: Power supply should be adequate for load. If fluctuation occurs, assist with 28 V battery to stabilize current.

5) Allow unit a 2 minute warm up.
6) Reset adjustment on alternator control unit for 1.0 A field current.
7) Decrease power supply (A) to approximately 28.6 V until ammeter shows 3 A field current.
8) Check alternator inoperative indicator by varying power supply (B) to 10.25 ± 0.75 V.



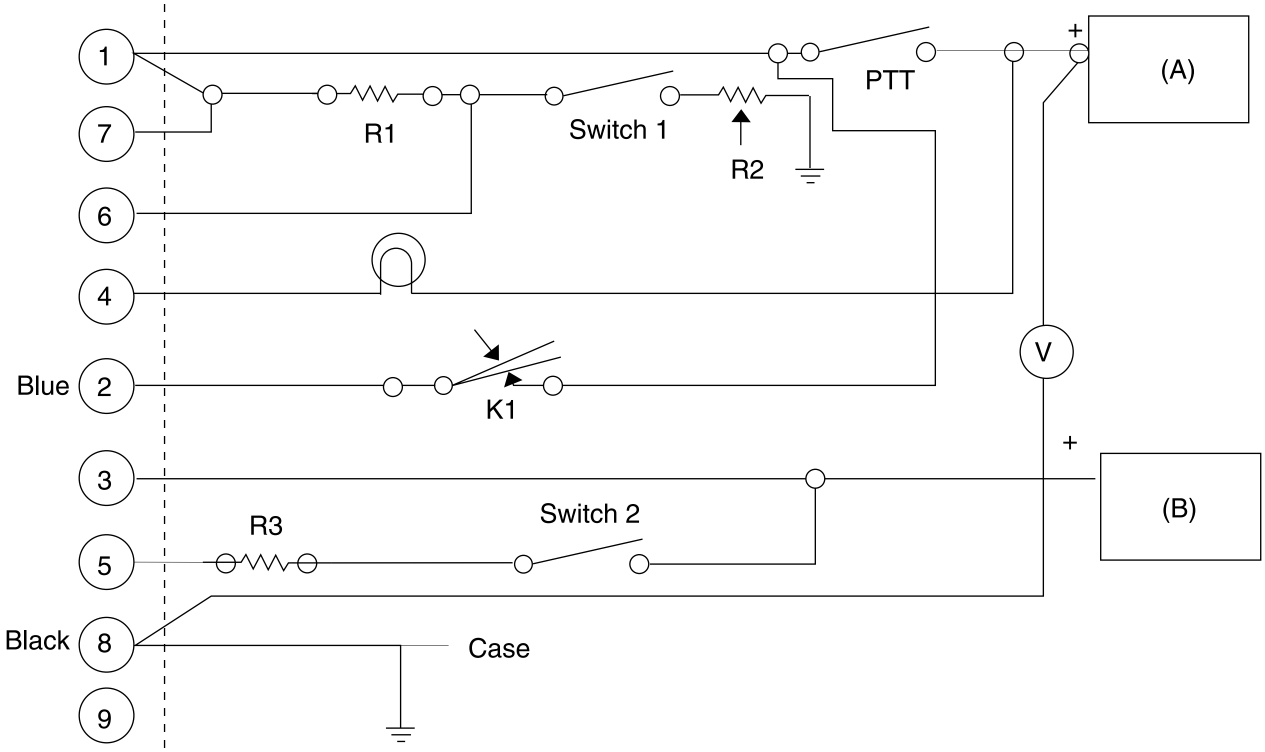
Bench Test of Alternator Control Unit - Lamar #B-00382-1
Figure 2

(c) Equalizer Test

1) Apply voltmeter at pins #5 and #8 (equalizer and ground). Equalizer voltage should read 5.75 V ± 0.1 V.

NOTE: Equalizer voltage outside 5.65 - 5.85 V limits may still be acceptable because of meter calibration differences and temperature. If error is several times the stated 0.1 V tolerance, have the unit rechecked by a fully equipped test facility.

2) Set R2 for maximum resistance (650 ž).
3) Close switch #1.
4) Adjust R2 for 50 MV across R1.
5) Read equalizer voltage.
6) Subtract from reading in step 1. Difference should be 10.0 V ± 0.25 V.



Overvoltage Test of Alternator Control Unit
Figure 3

(d) Overvoltage Protection Test (refer to "Figure 3")
  1) Revise connections as follows:
    e) Disconnect dummy load.
    f) Add 28 V relay (K1). No other connections should be on pin #2.
    g) Add a press-to-test (PTT) switch between power supply (A) and pin #1.
    h) Add resistor R3 and switch #2.
  2) Open switches #1 and #2. Set power supply (B) at 11 V.
  3) Set power supply (A) at 31.8 V. Depress PTT and hold for 5 seconds. No activation should occur.
  4) Increase power (A) to 32.2 V. Depress PTT. Relay should activate almost instantly.
  5) Close switch #2. Increase power to 33.8 V. Depress PTT for 5 seconds. No activation should occur.
  6) Increase power to 34.5 V. Depress PTT. Relay should activate almost instantly.

10. Battery (Dry-Charged Lead-Acid - Gill Model G-243)

S/N's 4636001 thru 4636374.

NOTE: Airplanes which have installed Kit 766-661, have a Gill Model G-247 increased amperage battery which is serviced the same as the Gill Model G-243 battery.

Service includes the entire battery system consisting of the battery, acid recovery jar, vents, and battery compartment. These services shall be completed every 50 operating hours or 30 days, whichever occurs first, and at every 100 hour inspection.

A. Troubleshooting

See "Chart 4".

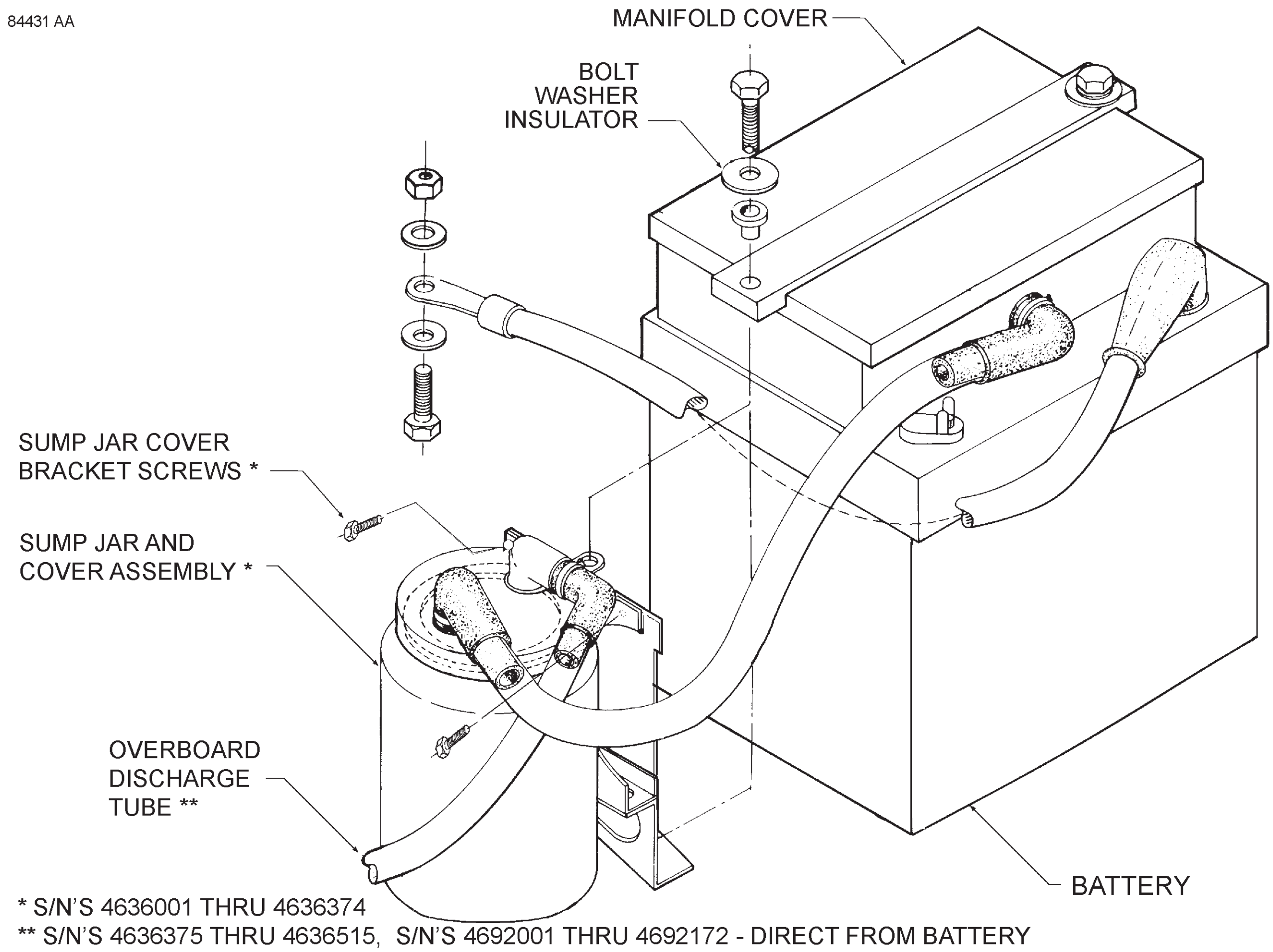
B. Removal

(1) Remove battery access plate from floor of forward baggage compartment.

CAUTION: REMOVE NEGATIVE CABLE FIRST, BEFORE EXPOSING AND LOOSENING POSITIVE TERMINAL WING NUT, TO AVOID SHORT CIRCUITS AND ELECTRICAL BURNS.

(2) Pull back rubber battery terminal boots, loosen wing nuts and disconnect battery cables.

(3) Remove vent hose at battery manifold.



Battery System Installation
Figure 4

CHART 4 (Sheet 1 of 2)
TROUBLESHOOTING (BATTERY SYSTEM)

Trouble	Cause	Remedy
Battery installed with reversed polarity.	Battery charged backwards.	Remove battery. Connect load, such as landing light lamp or similar load, and discharge battery. Recharge with correct polarity and test each cell for signs of damage due to reversed charging. Reinstall with corrected polarity.
NOTE: This type of condition can only occur in a case where a discharged battery has been removed from the airplane and put on a charger with the polarity reversed. This reversal in polarity cannot occur in the airplane due to any fault in the alternator system.		
Battery will not hold charge.	Battery worn out.	Replace battery.
	Standing too long.	Remove and recharge battery if left in unused airplane one week or more.
	Equipment left “ON” accidentally.	Remove and recharge.
	Impurities in electrolyte.	Replace battery, clean and recharge acid recovery jar.
	Short circuit (ground) in wiring.	Check wiring.
	Broken cell partitions.	Replace battery, clean and recharge acid recovery jar.
Battery life is short.	Overcharge due to level of electrolyte being below tops of plates.	Maintain electrolyte level. Clean and recharge acid recovery jar.
	Heavy discharge.	Replace battery, clean and recharge acid recovery jar.
	Sulfation due to disuse.	Replace battery, clean and recharge acid recovery jar.
	Impurities in electrolyte.	Replace battery, clean and recharge acid recovery jar.
Cracked cell.	Hold down was loose.	Replace battery and tighten, clean and recharge acid recovery jar.
	Frozen battery.	Replace battery, clean and recharge acid recovery jar.
Electrolyte runs out of vent plugs.	Too much water added to battery.	Drain and keep at proper level. Clean and recharge battery, vents and acid recovery jar.

CHART 4 (Sheet 2 of 2)
TROUBLESHOOTING (BATTERY SYSTEM)

Trouble	Cause	Remedy
Excessive corrosion inside container.	Spillage from over-fillings.	Use care in adding water.
	Vent lines leaking or clogged.	Replace vent lines. Also, clean and recharge acid recovery jar.
Battery freezes.	Discharged battery.	Replace battery, clean and recharge acid recovery jar.
	Water added and battery not charged immediately.	Always recharge battery at least 1/2 hour when adding water in freezing weather.
	Leaking cell.	Replace battery. Clean and recharge acid recovery jar.
Battery polarity reversed.	Connected backwards on airplane or charger.	Battery should be slowly discharged completely and then charged correctly and tested.
Battery consumes excessive water.	Cracked jar (one cell only).	Replace battery. Clean and recharge acid recovery jar.
Battery Disconnect Solenoid does not operate.	Open circuit.	Repair wiring.
	Open-circuited solenoid coil.	Replace unit.
Battery Disconnect Solenoid intermittent operation.	Short-circuited coil.	Replace coil.
	Loose electrical connection.	Clean and tighten electrical connections.
	Badly burned points.	If points cannot be dressed down, replace the unit.

(4) Remove battery hold-down bolts. Lift battery manifold cover off battery and place cover away from carpeting etc., due to acid contamination on inside surface.

(5) Lift battery out of compartment.

(6) Remove the vent hose that connects through the firewall from the acid recovery jar cover.

(7) Remove the two screws securing acid recovery jar cover to bracket. Slide the acid jar assembly upward off of the battery hold-down bar. Keep the jar upright in a vertical position.

C. Installation

(1) Keeping the jar upright in a vertical position, slide the acid jar assembly and bracket downward on to the battery hold-down bar. Secure recovery jar cover to bracket with the two screws previously removed.

(2) Install the vent hose that connects through the firewall at the acid recovery jar end.

(3) Place the battery into the compartment.

(4) Place battery manifold cover on battery and install battery hold down bolts.

(5) Install vent hose at battery manifold.

(6) Push rubber battery terminal boots back on, tighten wing nuts and connect battery cables.

(7) Install battery access plate to floor of forward baggage compartment.

D. Cleaning

(1) Battery

(a) Remove all accumulated contamination from the battery exterior with a stiff bristle brush. (Do not use a metal brush or abrasive materials.) Wipe exterior of battery and interior of manifold, including manifold top cover, with a cloth saturated with a solution of bicarbonate of soda mixed - one part soda to twenty parts of water. (Check that cell plugs are tight - do not allow soda solution to enter any cells.)

(b) Wash entire battery with clear water and dry thoroughly.

(c) Wash down the battery support and floor area, hold down supports, connectors and cable ends with a soda solution followed by clear water. Dry entire area and component parts thoroughly. Apply fresh acid-resistant paint if required.

(2) Acid Recovery Jar and Vent Lines

(a) Visually inspect all vent lines for kinks, cracks, flexibility, and loose connections. Replace only with special hoses from parts manual. (DO NOT REPLACE WITH ORDINARY RUBBER HOSE.)

(b) Slowly pour the soda solution into the vent hose, still attached to the firewall fitting, using a small funnel. The solution will flow out the bottom fuselage vent.

(c) Follow with a final purge of clear water to flush the vent line and then blow dry with low pressure air. This ensures that the vent line is not kinked or restricted and that it is neutralized.

(d) Wipe down the lower fuselage area surrounding the vent with soda solution and clear water. Apply a fresh coat of high quality aircraft wax to entire area.

(e) Unscrew the bottom of the recovery jar and separate from the top. Remove jar pad and empty jar contents into suitable container for safe disposal.

(f) Thoroughly wash and neutralize the jar, pad, top (including bracket), and the short length of vent hose still attached to the jar top with a soda solution and a clear water rinse.

(g) Thoroughly dry all components and recharge the jar with 0.75 of bicarbonate of soda. Place dry jar pad in the jar, on top of the soda charge.

(h) Screw jar back together and keep it in a vertical position.

(i) Install in aircraft per preceding installation instructions.

E. Charging - (Gill Model G-243)

CAUTION: WEAR EYE PROTECTION WHEN CHARGING A BATTERY - DO NOT TAKE CHANCES. MAKE SURE THE CHARGING AREA IS WELL VENTILATED. IF CENTRAL AIR CONDITIONING IS USED, THE BATTERY CHARGING AREA SHOULD BE VENTED TO THE OUTSIDE AIR TO PREVENT HYDROGEN GASES FROM BEING CIRCULATED THROUGHOUT THE BUILDING.

The National Electric Code forbids charging batteries installed in aircraft or within 10 feet of fuel tank areas. The battery must be removed from the aircraft for charging. Further, an aircraft battery should not be allowed to deteriorate to the point where safety of flight is jeopardized. The battery's emergency capacity should be sufficient to power the essential bus for a minimum of thirty minutes.

(1) Remove cell plugs and ensure that vents in plugs are open and that vent valves operate freely.

(2) Check that the electrolyte level in each cell is at the bottom of the split ring.

(3) A hydrometer check of each cell should be accomplished. (See Hydrometer Reading and Battery Charge, below.)

(4) It is recommended that vent caps be left on the battery while charging. In addition a wet cloth should be placed over the vent caps within the manifold.

(5) The battery may be charged at any rate, in amperes, not to exceed that point which would produce bubbling and gassing of the electrolyte or a cell temperature of 115° F in any case.

NOTE: If a cell temperature reaches the 115° F. limit, the charging rate shall be reduced and the charge completed at 3 amperes or less. DO NOT CHARGE AT A HIGHER RATE WHEN CELLS ARE GASSING. Refer to Gill Service Manual G.S.M.-682 for alternate charging methods and service procedures.

CHART 5
SPECIFIC GRAVITY TEMPERATURE CORRECTION

Electrolyte Temperature °C	°F	Correction	
60	140	1.024	
55	130	1.020	
49	120	1.016	**Add To**
43	110	.012	**Reading**
38	100	.008	
33	90	.004	
27	80	.000	
23	70	-.004	
15	60	-.008	
10	50	-.012	
5	40	-.016	**Subtract**
-2	30	-.020	**From**
-7	20	-.024	**Reading**
-13	10	-.028	
-18	0	-.032	
-23	-10	-.036	
-28	-20	-.040	
-35	-30-	-.044	

(6) If a constant current (recommended) charge is available, the charge should be started at 3 amperes and reduced in half if and when cells start gassing until fully charged.

(7) As charging occurs, if any cells sputter or flood, the electrolyte level is too high and the excess must be removed. In any case the electrolyte level shall be adjusted at the end of the charge. The level will rise due to acid returning to the electrolyte mix, normal gassing, and expansion due to temperature rise.

(8) Thoroughly clean battery after charging to prevent remaining acid bridges which can form during charging.

NOTE: Never allow lead acid batteries or tools used on them to be near Ni-Cad batteries and Ni-Cad battery tools.

F. Hydrometer Reading and Battery Charge

Whenever checking the battery, ascertain that all connections are clean and tight and that the fluid level is above the baffle plates. If it is necessary to add fluid, fill cell with distilled water to the bottom of the split ring. After adding water, charge the battery until gassing before taking a hydrometer reading. Otherwise, the water and electrolyte will not be mixed, giving a false reading. Temperatures different from the established norm will effect the hydrometer readings. Refer to "Chart 5" for the temperature corrections. Specific gravity values for a fully charged battery are shown in "Chart 6".

To adjust low specific gravity, charge the battery (see Charging, above) until it is gassing and until the specific gravity rises no higher over a 3-hour period. Then remove some electrolyte and replace with 1.300 specific gravity electrolyte. Repeat this step if, after one hour of charging, the specific gravity is still too low. DO NOT ADJUST A CELL THAT DOES NOT GAS.

CAUTION: IN THE OPERATION OF THE BATTERY, GASES ARE FORMED WHICH MAY BE EXPLOSIVE IF IGNITED. NEVER CREATE SPARKS OF ANY KIND OR BRING AN OPEN FLAME NEAR THE BATTERY. VENTILATE THE BATTERY WHEN CHARGING TO DISPOSE OF THE GAS GENERATED BY THE BATTERY.

To adjust high specific gravity, charge the battery (see Charging Battery) until it is gassing and until the specific gravity rises no higher over a 3-hour period. Remove some electrolyte and replace with distilled water. Repeat this step if, after one hour of charging, the specific gravity is still too high.

CHART 6
ELECTROLYTE TEMPERATURE CORRECTIONS

Electrolyte Temperature	Specific Gravity
47° F	1.280 to 1.300
77° F	1.280 to 1.290
107° F	1.260 to 1.280
Temperature change of 30° F changes the reading 0.010.	

CHART 7
DISCHARGE RATES

TELEDYNE Battery Type	Volts	(20 HRS.) Ampere Hours
GILL-G-243	24	16
GILL-G-247	24	32

G. Discharge

The capacity of a storage battery is measured in units of ampere hours, which is the product of the electrical current in amperes multiplied by the time in hours. Although current may be obtained after the end of the time, the voltage of the battery has dropped to a point beyond which it is not very useful. The ampere hours which may be obtained from a battery are greater for a long low-rate or intermittent rate discharge than for a short high-rate discharge because the voltage will drop faster at the higher discharge rate. The maximum permissible rate of discharge is limited only by the current-carrying ability of the wiring, motor, or other apparatus to which the battery is connected or by the current-carrying ability of the cell terminals and connectors and not by the plates themselves. Recommended discharge rates are listed in “Chart 7”.

H. Temperature Considerations

Operation of storage batteries beyond their ambient temperature or charging voltage limits will result in excessive cell temperatures leading to electrolyte boiling, rapid deterioration of the cell, and finally, battery failure. The relationship between the maximum charging voltage and the number of cells in the battery is also significant, since this will determine (for a given ambient temperature and state of charge) the rate at which energy is absorbed as heat within the battery. The maximum voltage per cell should not exceed 2.35 - volts, and the maximum temperature should not exceed 115° F.

Low electrolyte temperatures temporarily reduce the battery capacity and the freezing point depends on the specific gravity. To prevent freeze damage, maintain the specific gravity at a reasonably high level as indicated by “Chart 8”.

NOTE: Lead-acid batteries are subject to a constant discharge due to the internal chemical action.

CHART 8
ELECTROLYTE FREEZING POINTS

Specific Gravity	Freezing Point °C	°F
1.300	-70	-95
1.275	-62	-80
1.250	-52	-62
1.225	-37	-35
1.200	-26	-16
1.175	-20	-4
1.150	-15	5
1.125	-10	13
1.100	-8	19

I. Repairs, Storage, and Service Tips

The internal parts of the battery have been designed to wear at approximately the same rate, making it uneconomical to replace any of the parts with new ones. Replacing the entire battery is simpler and cheaper.

Before storing the battery, it should be properly charged, the vent plugs put tightly in place, and the leads disconnected to prevent use during idle periods. The battery should be charged at intervals during the idle period. Before returning the battery to service, it should be thoroughly charged. The battery will be sufficiently charged when, after a 3-hour period, the specific gravity does not rise any higher with the electrolyte gassing and a charging rate of 1-1/2 amperes.

Long battery life and trouble-free service is obtained from the battery if the following simple tips are observed:

(1) Keep it clean.

(2) Keep it charged.

(3) Maintain proper electrolyte levels.

(4) Keep specific gravity equal among all cells.

J. Preparing New Dry-Charged Battery for Installation

CAUTION: PRIOR TO INSTALLING A NEW DRY-CHARGED BATTERY, FOLLOW THE PREPARATION INSTALLATION INSTRUCTIONS FURNISHED WITH THE BATTERY BY TELEDYNE.

The Teledyne, dry-charged, Model GILL/G-243 Battery shall be stored as received from Teledyne. Do not remove vent seals, add acid, nor attempt to charge a dry-charged battery until the time arrives to install the battery into an airplane.

11. Battery (Valve-Regulated Lead-Acid - Gill Model G-250S)

(S/N's 4636375 thru 4636515) (S/N's 4692001 thru 4692172)

Gill valve-regulated lead-acid (VRLA) non-maintainable batteries are fully charged when they leave the factory. Service includes the entire battery system consisting of the battery and battery compartment. These services shall be completed every 50 operating hours or 30 days, whichever occurs first, and at every 100 hour inspection.

A. Troubleshooting

See "Chart 4" on page 243019.

B. Removal

(1) Remove battery access plate from floor of forward baggage compartment.

CAUTION: REMOVE NEGATIVE CABLE FIRST, BEFORE EXPOSING AND LOOSENING POSITIVE TERMINAL WING NUT, TO AVOID SHORT CIRCUITS AND ELECTRICAL BURNS.

(2) Pull back rubber battery terminal boots, loosen wing nuts and disconnect battery cables.

(3) Remove vent hose at battery manifold.

(4) Remove battery hold down bolts. Lift battery manifold cover off battery and place cover away from carpeting etc, due to acid contamination on inside surface.

(5) Lift battery out of compartment.

(6) Remove the vent hose that connects through the firewall.

C. Installation

(1) Install the vent hose that connects through the firewall.

(2) Place the battery into the compartment.

(3) Place battery manifold cover on battery and install battery hold-down bolts.

(4) Install vent hose at battery manifold.

(5) Connect battery cables, tighten wing nuts and push rubber battery terminal boots back on.

(6) Install battery access plate to floor of forward baggage compartment.

D. Cleaning

(1) Battery

(a) Remove all accumulated contamination from the battery exterior with a stiff bristle brush. (Do not use a metal brush or abrasive materials.) Wipe exterior of battery and interior of manifold, including manifold top cover, with a cloth saturated with a solution of bicarbonate of soda mixed - one part soda to twenty parts of water. (Check that vent caps are sealed in place - At no time must these vents caps be removed.)

(b) Wash entire battery with clear water and dry thoroughly.

(c) Wash down the battery support and floor area, hold-down supports, connectors and cable ends with a soda solution followed by clear water. Dry entire area and component parts thoroughly. Apply fresh, acid-resistant paint, if required.

E. Charging - (Gill Model G-250S)

CAUTION: WEAR EYE PROTECTION WHEN CHARGING A BATTERY - DO NOT TAKE CHANCES. MAKE SURE THE CHARGING AREA IS WELL VENTILATED. IF CENTRAL AIR CONDITIONING IS USED, THE BATTERY CHARGING AREA SHOULD BE VENTED TO THE OUTSIDE AIR TO PREVENT HYDROGEN GASES FROM BEING CIRCULATED THROUGHOUT THE BUILDING.

The National Electric Code forbids charging batteries installed in aircraft or within 10 feet of fuel tank areas. The battery must be removed from the aircraft for charging. Further, an aircraft battery should not be allowed to deteriorate to the point where safety of flight is jeopardized. The batteries emergency capacity should be sufficient to power the essential bus for a minimum of thirty minutes.

Check battery voltage and date of manufacture prior to installation. If the open circuit voltage is less than 25.0 volts, or if 90 days has passed since date of manufacture, the battery should be charged before placing into service.

(1) If the battery voltage is below 25.0 volts, the battery should be recharged.

(2) Remove the battery from the aircraft and recharge in a well-ventilated area. To avoid water loss and possible overcharge, the manufacturer recommends a constant potential (tapering) charge.

NOTE: Non-maintainable means that this type of battery cannot be opened to add fluid. Periodic maintenance is still required in the form of boost charging if the battery becomes discharged.

(3) CONSTANT POTENTIAL (TAPERING) CHARGE: Charge the battery at 28.2 to 28.6 volts for a period of 24 hours. Inrush current can be as high as 100 amps, provided voltage does not exceed as mentioned above.

(4) CONSTANT CURRENT CHARGE: Charge the battery at rate listed (see “Chart 9”) for a period of 10 to 13 hours, unless otherwise stated, or until the battery voltage stabilizes for three (3) consecutive hours, or drops.

CHART 9
CONSTANT CURRENT CHARGE

TELEDYNE Battery Type	CONSTANT CURRENT CHARGE RATES (AMPS)	CHARGING RATES/CAPACITY 30 MIN EMERGENCY RATE (AMPS)	1 HOUR RATE (AMPS)
GILL-G250S	1.0	23	15.0

F. Battery Capacity Test - (Gill Model G-250S)

(1) Remove battery from aircraft.

(2) Charge it in compliance with charging instructions for sealed batteries. (See Charging - Gill Model G-250S, above.)

(3) Allow battery to stand on open circuit for one (1) hour.

(4) Connect fully charged battery to capacity tester that incorporates a load resistance, amp meter, voltmeter and a time clock (timer).

(5) Discharge battery at the 1 hour capacity rate to 1.67 volts per cell (20.0 volts). Note the discharge time to the end voltage.

The battery is considered airworthy if it meets 80% of the 1 hour capacity rating (minimum of 48 minutes to the cut-off voltage).

If the battery fails to meet the minimum run time, recharge using the constant potential method until charge current stabilizes (this could take 10 to 24 hours). Allow the battery to stand on open circuit for one (1) hour after recharging. Repeat the above capacity test. If the failure persists, replace the battery.

If battery is found to be airworthy as noted above, recharge the battery prior to re-installing it in the aircraft.

12. Battery (Valve-Regulated Sealed Lead-Acid - Concorde Model RG24-17 / RG24-20)

The Concorde Battery RG24-17 is installed as standard equipment in S/N's 4636516–4636759 and 4692173 and up. Available with Kit P/N 88495-002, the Concorde Battery RG24-17 is installed in S/N's 4636001 thru 4636515 and S/N's 4692001 thru 4692172.

The Concorde Battery RG24-20 is installed as standard equipment in S/N's 4636760 and up.

Concorde valve-regulated sealed lead-acid (VRSLA) non-maintainable batteries are fully charged when they leave the factory. Service includes the entire battery system consisting of the battery and battery compartment. These services shall be completed every 50 operating hours or 30 days, whichever occurs first, and at every 100 hour inspection.

Service and maintain these batteries per the appropriate Concorde Component Maintenance Manuals (CMM). See Vendor Publications under Supplementary Publications in the Introduction.

A. Troubleshooting

See "Chart 10" on page 243028.

B. Definitions

(1) Valve Regulated lead-acid (VRLA) battery - A lead-acid battery in which there is no free electrolyte and the internal pressure is regulated by a pressure relief valve. This battery requires no maintenance of the liquid level and recombines the gases formed on charge within the battery to reform water. The battery may be used in any attitude without danger of leakage or spilling of electrolyte.

CHART 10
TROUBLESHOOTING (BATTERY SYSTEM)

Trouble	Cause	Remedy
Low Voltage or No Voltage.	Battery Partially or Fully Discharged.	Perform capacity test.
	Battery fuse blown or circuit breaker tripped.	Inspect and replace fuse or reset circuit breaker.
Battery capacity less than 85% of rating	Battery beyond serviceable life.	Replace Battery.
Battery does not hold charge.	Battery beyond serviceable life.	Replace Battery.
Battery gets hot while recharging.	Battery beyond serviceable life.	Replace Battery.

(2) Rated Capacity (C_1) - Quantity of electricity in Ampere-hours (Ah) which the cell or battery is capable of delivering in 1 hour, throughout its normal service life, after full charge, under conditions defined with regard to temperature and end discharge voltage.

(3) C_1 Rate - The rate, in amperes, equal to the battery's rated C_1 capacity. For example, the C_1 rate of a battery rated at 3.5Ah is 3.5 amperes.

(4) End Point Voltage (EPV) - The voltage at which the discharge current is terminated when measuring battery capacity. Unless otherwise stated, the EPV is equal to 20.0 volts for 24 volt batteries.

(5) Open Circuit Voltage (OCV) - The voltage of the battery at rest (no charging or discharging current present). A stable OCV requires a rest of at least four hours.

C. Precautions:

WARNING: LOW CAPACITY HAZARD. AIRCRAFT BATTERIES ARE CERTIFIED TO HAVE CERTAIN MINIMUM CAPACITY FOR EMERGENCY OPERATIONS IN THE EVENT OF A ELECTRICAL GENERATOR SYSTEM FAILURE. NEVER USE A BATTERY THAT HAS LESS THAN 80% OF RATED CAPACITY.

WARNING: BATTERIES ON CHARGE OR DISCHARGE PRODUCE HYDROGEN GAS, WHICH CAN EXPLODE IF IGNITED. DO NOT SMOKE, USE AN OPEN FLAME, OR CAUSE SPARKING NEAR A BATTERY. CHARGE, SERVICE OR TEST THE BATTERY ONLY IN A WELL VENTILATED AREA. THE USE OF EXHAUST FANS MAY REDUCE THE RISK OF EXPLOSION.

WARNING: BATTERIES CONTAIN SULFURIC ACID WHICH WILL CAUSE BURNS. DO NOT TOUCH EYES AFTER TOUCHING BATTERY. DO NOT GET ACID IN YOUR EYES, ON YOUR SKIN, OR IN YOUR CLOTHING. IN THE EVENT OF ACID IN THE EYES, FLUSH THOROUGHLY WITH CLEAN COOL WATER FOR SEVERAL MINUTES. GET PROFESSIONAL MEDICAL ATTENTION. REFER TO BATTERY MATERIAL SAFETY DATA SHEET (MSDS) FOR ADDITIONAL INFORMATION.

WARNING: ONLY CONSTANT POTENTIAL CHARGING MAY BE DONE ON THE AIRCRAFT. DO NOT PERFORM A CONDITIONING CHARGE WHILE BATTERY IS INSTALLED ON THE AIRCRAFT. THERE MAY BE A SERIOUS RISK OF INJURY TO PERSONNEL AND / OR DAMAGE TO THE AIRCRAFT OR AIRCRAFT SYSTEMS DUE TO HIGH VOLTAGE AND GENERATION OF EXPLOSIVE GASES WHEN CHARGING CONSTANT CURRENT.

WARNING: WEAR PROPER EYE, FACE, AND HAND PROTECTION AT ALL TIMES WHEN WORKING WITH BATTERIES. KNOW THE LOCATION AND USE OF EMERGENCY EYEWASH AND SHOWER NEAREST THE BATTERY CHARGING AREA.

WARNING: ELECTRIC SHOCK HAZARD. DO NOT TOUCH UNINSULATED PORTION OF THE CONNECTOR OR THE BATTERY TERMINALS. A POSSIBILITY OF SERIOUS ELECTRICAL SHOCK EXISTS

WARNING: ELECTRIC SHOCK HAZARD. DO NOT LAY TOOLS OR OTHER METAL OBJECTS ON THE BATTERY AS ARCING OR EXPLOSION COULD OCCUR. REMOVE CONDUCTIVE JEWELRY BEFORE WORKING AROUND BATTERY, CHARGER, OR TEST EQUIPMENT.

CAUTION: ELECTRIC BURN HAZARD. DO NOT WEAR CONDUCTIVE RINGS, BELT BUCKLES, OR OTHER JEWELRY WHEN WORKING WITH BATTERIES, CHARGERS, OR TEST EQUIPMENT. DO NOT LAY TOOLS OR OTHER METAL OBJECTS ON THE BATTERY AS ARCING AND SEVERE BURNS COULD OCCUR.

CAUTION: TO PREVENT DAMAGE TO THE CONNECTOR, ARC BURNS, OR EXPLOSION, BATTERIES SHOULD NEVER BE CONNECTED OR DISCONNECTED WHILE BEING CHARGED OR DISCHARGED. BATTERIES MUST BE CONNECTED OR DISCONNECTED ONLY WHEN THE CIRCUIT IS OPEN. ENSURE THE AIRCRAFT BATTERY SWITCH, EXTERNAL POWER SOURCE, OR THE CHARGER/ANALYZER IS IN THE “OFF” POSITION BEFORE CONNECTING OR DISCONNECTING THE BATTERY. BATTERY TERMINAL PROTECTORS SHOULD BE INSTALLED WHENEVER THE BATTERY IS NOT CONNECTED IN THE AIRCRAFT OR TO THE TEST EQUIPMENT.

CAUTION: BATTERIES CONTAIN HAZARDOUS MATERIALS. KNOW THE LOCATION AND PROPER USE OF EMERGENCY RESPONSE MATERIALS. REFER TO BATTERY MATERIAL SAFETY DATA SHEET (MSDS) FOR ADDITIONAL INFORMATION.

D. Removal

(1) Remove battery access plate from floor of forward baggage compartment.

CAUTION: REMOVE NEGATIVE CABLE FIRST, BEFORE EXPOSING AND LOOSENING POSITIVE TERMINAL NUT, TO AVOID SHORT CIRCUITS AND ELECTRICAL BURNS.

(2) Pull back rubber battery terminal boot, loosen nuts and disconnect battery cables.

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(3) Remove battery hold down bolts. Lift battery strap off battery and place away from carpeting, etc.

(4) Lift battery out of compartment.

E. Installation

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(1) Place the battery into the compartment.

(2) Place battery strap on battery and install battery hold-down bolts. Torque per “Figure 5” on page 243030.

(3) Connect battery cables, tighten nuts and push rubber battery terminal boot back on.

(4) Install battery access plate to floor of forward baggage compartment.

F. Cleaning

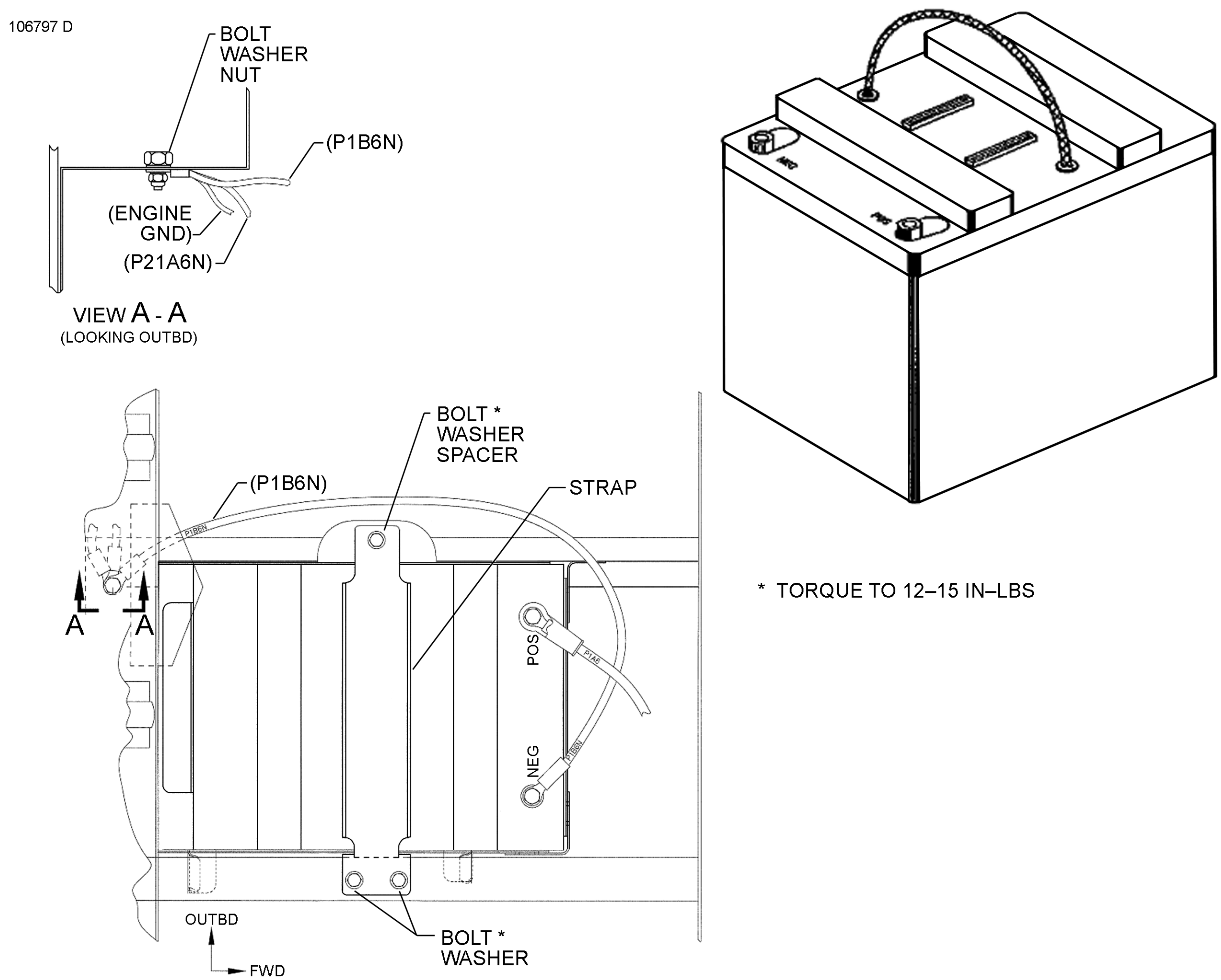
(1) Battery

CAUTION: DO NOT USE ANY TYPE OF SOLVENT TO CLEAN THE BATTERY. SOLVENTS MAY DAMAGE THE BATTERY.

(a) Clean the outside surfaces of the battery with a lint-free shop cloth that is clean, dry, and free of oil.

(b) If the battery has caked-on dirt or grime, use a cloth dampened with tap water, then wipe dry.

(c) Wipe down the battery support and floor area, hold-down supports, connectors and cable ends.



Concorde RG24-17 / RG24-20 Battery System Installation
Figure 5

G. Charging - (Concorde Model RG24-17 / RG24-20)

CAUTION: WEAR EYE PROTECTION WHEN CHARGING A BATTERY - DO NOT TAKE CHANCES. MAKE SURE THE CHARGING AREA IS WELL VENTILATED. IF CENTRAL AIR CONDITIONING IS USED, THE BATTERY CHARGING AREA SHOULD BE VENTED TO THE OUTSIDE AIR TO PREVENT HYDROGEN GASES FROM BEING CIRCULATED THROUGHOUT THE BUILDING.

The National Electric Code forbids charging batteries installed in aircraft or within 10 feet of fuel tank areas. The battery must be removed from the aircraft for charging. Further, an aircraft battery should not be allowed to deteriorate to the point where safety of flight is jeopardized. The batteries emergency capacity should be sufficient to power the essential bus for a minimum of thirty minutes.

Check battery voltage and date of manufacture prior to installation. If the open circuit voltage is less than 25.0 volts, or if 90 days has passed since date of manufacture, the battery should be charged before placing into service.

NOTE: If a battery gets very hot (external case temperature greater than 55°C (130°F) during this charge, stop the current and allow the battery to cool to room temperature before continuing the charge.

(1) If the battery voltage is below 20.0 volts, the battery should be recharged.

(2) Remove the battery from the aircraft and recharge in a well-ventilated area.

NOTE: Non-maintainable means that this type of battery cannot be opened to add fluid. Periodic maintenance is still required in the form of boost charging if the battery becomes discharged.

(3) Constant Potential (Tapering) Charge:

NOTE: Constant Potential Charging is the preferred method of charging the battery. If the battery does not pass the capacity test, the Conditioning Charge Procedure should then be used.

(a) Connect the battery terminals to the constant potential charging equipment.

(b) Apply a constant potential of 28.25 ± 0.25 volts (for 24 volt batteries) with a current capability of at least C1 amperes.

(c) Continue charging until the charge current stabilizes (within 10%) for three consecutive hourly readings.

(d) Allow the battery to cool down for at least one hour before any other tests are performed.

(4) Conditioning Charge Procedure:

WARNING: CHARGE THE BATTERY ONLY IN A WELL VENTILATED AREA BECAUSE A SIGNIFICANT AMOUNT OF HYDROGEN GAS MAY BE RELEASED FROM THE BATTERY.

(a) Discharge the battery at the C1 rate to the EPV. If battery has already been discharged to the EPV, skip this step.

(b) Connect the battery terminals to the constant current charging equipment.

(c) Charge at a constant current rate of C1/10 until the voltage on charge is 31.0 volts or higher for four (4) hours, then discontinue charging.

NOTE: If a battery gets very hot (external case temperature greater than 55°C (130°F) during this charge, stop the current and allow the battery to cool to room temperature before continuing the charge.

(d) After charging, allow the battery to cool down for at least eight (8) hours before any other tests are performed.

H. Battery Capacity Test - (Concorde Model RG24-17 / RG24-20))

(1) Special Tools:

CAUTION: CONTACT CONCORDE FOR DETERMINATION OF EQUIVALENT TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY THE BATTERY.

(2) Test Procedure:

(a) Stabilize the battery at 20° C (68° F) or higher. The battery must be at this temperature for at least 20 hours.

(b) Depending on the type of charger available, charge the battery Constant Potential (CP). See Constant Potential (Tapering) Charge, above.

(c) Connect the battery to the discharge equipment and discharge at the C1 rate on the label.

(d) Discharge the battery to the EPV.

(e) Record the time to EPV. The battery is acceptable for continued use if the ampere hour capacity (actual hours of discharge x ampere rate of discharge) is 51 minutes or greater (85% of rated C1 capacity or greater).

1) If the battery passes the capacity test, see Return to Service, below.

2) If the battery fails the capacity test, perform the conditioning procedure under Conditioning Charge Procedure, above. After the battery has been conditioned, perform a second capacity test.

3) If the battery passes the second capacity test, return it to service. If the battery fails the second capacity test, repeat the conditioning charge and repeat the capacity test.

4) If the battery passes the third capacity test, return it to service. If the battery fails the third capacity test, the battery should be replaced.

NOTE: If a battery gets very hot (external case temperature greater than 55°C (130°F) during charging, replace the battery.

I. Return to Service

(1) Charge the battery per Battery Charging, see “Charging - (Concorde Model RG24-17 / RG24-20)” on page 243031.

(2) Install the battery per Installation, see “Installation” on page 243029.

EXTERNAL POWER

1. Emergency Starting through External Power Receptacle

CAUTION: DO NOT CONNECT EXTERNAL POWER PLUG TO AIRCRAFT'S EXTERNAL POWER RECEPTACLE WITH A DISCHARGED OR "RUN-DOWN" AIRCRAFT BATTERY, UNLESS THE AIRCRAFT'S BATTERY MASTER SWITCH IS TURNED "OFF" AND LEFT "OFF".

A. Place ground power unit on the pilot's side of the nose in full view of pilot's window. Upon connecting ground power to the aircraft, the External Power Contactor will energize immediately, supplying electrical power to all aircraft electrical buses.

B. With Battery Master Switch left in the "OFF" position, turn "OFF" all unnecessary electrical loads as in a normal start.

CAUTION: EXERCISE GREAT CARE DISCONNECTING GROUND POWER PLUG. DISCONNECT THE PLUG ONLY FROM THE PILOT'S SIDE OF THE AIRPLANE.

C. Start engine and move ground power unit well away from aircraft.

D. Turn "ON" Battery Master Switch.

E. Observe Ammeter indicates Alternator electrical power is available.

F. With the alternator on-line, observe charging current on ammeters.

G. Do not take-off until charging current falls below 20 AMPS.

NOTE: The aircraft battery must be removed from the airplane if it is to be charged with a ground power dc supply.

NOTE: The Ammeters are displayed on the MFD when either Avidyne Entegra or Garmin G1000/ G1000 NXi is installed.

CHART 1
CAPACITY RATINGS AT DISCHARGE RATES

Battery = Teledyne/Gill G243 (24 Volts)		
Capacity Ratings		
1 Hour (Amp Hours)	**Emergency** (30 minutes) (Amps)	**Cold Cranking** (60 Seconds) (Amp @ 0°F)
10	17	150

Definitions

1 Hour:	The ampere hours shown divided by one hour is the rate of discharge to 1.5-volts per cell.
Emergency:	The amperes shown is the rate of discharge for 30 minutes to a cut-off voltage of 1.75-volts per cell.
Cold Cranking:	The amperes shown in discharge current used to crank an engine for 30 seconds at 0° F to 1.2-volts per cell.

2. Ground Power through External Power Receptacle

CAUTION: WHEN USING GROUND POWER UNIT, ENSURE THE VOLTAGE AND POLARITY ARE SET TO CORRESPOND WITH THE AIRCRAFT SYSTEM VOLTAGE AND POLARITY

A. Turn Battery Master Switch “OFF”, turn “OFF” all electrical equipment.

B. Place ground power unit on the pilot’s side of the nose in full view of pilot’s window. Upon connecting ground power to the aircraft, the External Power Contactor will energize immediately, supplying electrical power to all aircraft electrical buses.

C. Turn “ON” Battery Master Switch and turn “ON” electrical equipment to be used during ground operation.

CAUTION: EXERCISE GREAT CARE DISCONNECTING GROUND POWER PLUG. DISCONNECT THE PLUG ONLY FROM THE PILOT’S SIDE OF THE AIRPLANE.

D. When ground operation is complete, turn “OFF” all electrical equipment, turn Battery Master Switch “OFF” and disconnect ground power plug from aircraft.

ELECTRICAL LOAD DISTRIBUTION

1. Emergency Operation

NOTE: All equipment will be operational with only one alternator functioning. To prevent overload with only one alternator functioning, turn off non-essential equipment and any other equipment necessary to reduce the load to less than 70 amps (S/N's 4636001 thru 4636131) or 75 amps (S/N's 4636132 and up) (S/N's 4692001 and up).

CAUTION: DO NOT MANUALLY OPERATE (PULL) THE ALTR1 OR ALTR2 CIRCUIT BREAKER WHEN THE ENGINE IS RUNNING AND THE ALTERNATOR IS OPERATING PROPERLY.

A. With engine operating, turn Alternator No. 2 OFF. Check the appropriate ALTERNATOR INOP light is illuminated. On G1000/G1000 NXi equipped aircraft, when you turn OFF an alternator, the "ALT FAIL" CAS message will only be displayed if the alternator is ON and it has failed.

CAUTION: SHOULD THE LOW BUS VOLTAGE LIGHT ILLUMINATE WHILE ENGAGING ELECTRICAL EQUIPMENT, IMMEDIATELY REDUCE ELECTRICAL LOAD TO LESS THAN:

-- 70 AMPS in S/N's 4636001 THRU 4636131, OR

-- 75 AMPS in S/N's 4636132 AND UP and in S/N's 4692001 AND UP

B. Turn ON essential electrical equipment, one unit at a time. Essential electrical equipment may consist of:

1. Instrument Panel Lights
2. Annunciator Lights
3. Landing Gear Warning Horn
4. Stall Detector
5. Fuel Management
6. Left and Right Fuel Pumps
7. Fuel Quantity
8. Electric Tachometer
9. Engine Gauges
10. Engine Monitor
11. Pilots Turn Coordinator
12. Landing Gear Indicator Lights
13. Hydraulic Pump Control
14. Hydraulic Pump Power
15. ***One*** Communication Transceiver
16. ***One*** Navigation Receiver
17. Speaker Amplifier
18. ***One*** Transponder
19. Compass
20. Avionics Cooling Fan
21. Any other equipment that may be ***essential*** for flight operations

C. Turn Alternator No. 2 ON; turn Alternator No. 1 OFF. Check to be sure that all equipment that was supported by the No. 1 Alternator is supported by the No. 2 Alternator.

D. If necessary, the non-essential bus, and its related equipment, can be eliminated by pulling the non-essential 70-amp circuit breaker on the Tie Bus Circuit Breaker panel.

E. Upon completing the foregoing checks, be sure that all electrical switches are OFF and all circuit breakers have been reset.

2. Electrical Load Analysis (ELA)

The following charts are provided to aid in determining available electrical capacity when adding or removing equipment over the life of the airplane.

A. PA-46-350P

(1) Mirage Standard or Avidyne Entegra equipped (1995–2010) (PIR-VB-1349, Rev. K.)

See “Chart 1” on page 24605.

(2) Mirage Garmin G1000 equipped (PIR-VB-1349, Rev. K.)

S/N’s 4636460, 4636463-4636651, less 4636481, 4636633
See “Chart 2” on page 24608.

(3) M350 Garmin G1000 equipped (PIR-VB-1349, Rev. N.)

S/N’s 4636633, 4636652 thru 4636715, 4636717–4636719
See “Chart 4” on page 246020.

(4) M350 Garmin G1000 NXi equipped (PIR-VB-1349, Rev. P.)

S/N’s 4636716, 4636720–4636775
See “Chart 5” on page 246027.

(5) M350 Garmin G1000 NXi equipped (PIR-VB-1349, Rev. R.)

S/N’s 4636776 and up
See “Chart 6” on page 246031.

B. PA-46R-350T

(1) Matrix Avidyne Entegra equipped (PIR-VB-1349, Rev. K.)

S/N’s 4692001–4692133, and 4692141, 4692149 and 4692153
See “Chart 1”.

(2) Matrix Garmin G1000 equipped with S-TEC Autopilot (PIR-VB-1999, Rev. A.)

S/N’s 4692134 and up less 4692141, 4692149, 4692153
See “Chart 3” on page 246014.

3. Volt / Ammeter

The volt / ammeter combination indicator (S/N’s 4636001 thru 4636374) consists of two ammeters and a volt meter mounted in one case. The ammeters indicate amperage load on each alternator. The voltmeter indicates total system voltage. The best troubleshooting approach to determine this instruments condition, is to compare the readings with a known good meter.

Should rapid and steady-state ammeter needle oscillation occur, first thoroughly clean all traces of corrosion from all grounding points throughout the electrical power distribution system. This is not to be confused with normal ammeter fluctuations showing changing current flow within the aircraft’s electrical system. The combined ammeter readings indicates the total current flow, except for the engine starter.

The voltmeter portion of the instrument shows main bus voltage. Should the instrument fail to indicate correctly, substitute a known good voltmeter before condemning the instrument. Placing a good voltmeter in parallel with the indicator is the best practice.

NOTE: The Ammeters are displayed on the MFD when either Avidyne Entegra or Garmin G1000/ G1000 NXi is installed.

4. USB Charging Ports

In S/N's 4636632, 4636652 and up, three dual USB charging ports are available. One is installed in the forward side of the main spar cover on the aircraft centerline to service the cockpit. The other two are installed in the left and right switch plates adjacent to the aft-facing cabin seats.

In S/N's 4636633, 4636652–4636715, 4636717–4636719, 4636760 and up, an optional USB charging port installation may have been installed. A single USB port is installed on each side of the cockpit and dual USB ports are installed in the left and right switch plates adjacent to the aft-facing cabin seats.

A. Description

The USB Charging Ports convert 28 VDC ship's current to standard 5V power for any electronic product that charges using a USB connector. The Universal Serial Bus-A (USB-A) port(s) at each location are designed as DCPs (Dedicated Charging Ports) to industry-standard protocol per the USB Battery Charging 1.2 Compliance Plan. Each port can provide up to 2.1 amps of power simultaneously to charge any USB device.

B. Operation

(1) Electrical Performance

The USB Charging Port converts an aircraft (DC) input voltage within the range specified to a 5V (DC) output. This output power is applied to a USB-A connector in accordance with the USB Implementers Forum.

The USB D+ and D- data lines communicate with the USB portable device to tell the device it is a dedicated charging port (DCP), capable of a higher current than a standard USB port. This allows the USB portable device to draw up to 2.1 Amps.

The unit is designed as a DC-to-DC converter with a series switch on each output to regulate current applied to that output. Each series switch independently reduces the output current to a safe level if the USB portable device draws excess current, is shorted or has a fault.

If the temperature of the unit becomes elevated due to a fault or excessive load, the device will seamlessly communicate with the USB portable device to lower the charge current. This allows the device to continue charging while the unit returns to a temperature within designed limits. When the temperature returns to a safe level the unit will automatically reestablish the higher charge current level with the device and continue charging.

(2) Protective Features

(a) Short Circuit Protection

The unit is capable of surviving a short circuit event without permanent damage. The unit goes into an over-current condition so that the average current is significantly reduced and the device is protected.

(b) Over-Current Protection

The unit monitors the current draw individually on each port. During an over-current condition the voltage is reduced. If the voltage falls below 3.8 VDC the output is turned off for a period of 12 seconds. The output is then checked for continued over-current conditions every 16 milliseconds. This condition is referred to as a hiccup mode. The device stays in this mode until the over-current condition is removed, then returns to normal operation.

(c) Low Input Voltage Shutdown

If the input voltage applied to the unit drops below 10 VDC the unit will shut down until the applied voltage returns to a level within range.

(d) Over-Temperature

When the temperature of the unit becomes elevated, the unit communicates with the USB portable device to reduce the charge current output (1 amp limit). When the temperature returns to an acceptable level the unit automatically returns to a higher charge current as required (up to 2.1 amps).

C. Troubleshooting

Troubleshooting of the USB Charging Ports is limited to verifying correct harness wiring: 10–28 VDC at Pin A and a good ground at Pin B; and that the harness is then connected to the unit. If the harness has been verified and is properly connected and energized and the unit fails to function properly - remove and replace the unit.

D. Removal

The cockpit unit is mounted to a bezel installed in the forward side of the main spar cover. The cabin units are mounted to the right and left switch plates adjacent to the aft-facing seats. All three units are removed and installed by the same procedure.

(1) Disconnect the negative (ground) battery cable.
(2) Remove the two screws securing the bezel or switch plate.
(3) Pull the bezel / switch plate out to expose the charging port unit and wiring harness(es).
(4) Disconnect and secure the charging port wiring harness.
(5) Remove the two screws securing the charging port unit to the bezel / switch plate and remove the unit.

E. Installation

The following assumes the bezel or switch plate has not been reinstalled since previous removal of the USB Charging Port unit.

(1) Position charging port unit behind the bezel / switch plate and secure with two screws.
(2) Reconnect wiring harness to the charging port.
(3) Place the bezel / switch plate in position and secure with two screws.
(4) Reconnect the negative (ground) battery cable.
(5) Turn ON the master switch.
(6) Connect a USB device to each charging port and verify charging.
    (a) If charging occurs at each port, install is complete.
    (b) If charging does not occur at one or both ports, troubleshoot per “Troubleshooting”, above.

CHART 1 (Sheet 1 of 3)
ELECTRICAL LOAD ANALYSIS (PA-46-350P 1995–2010)

Bus	Component / System	Electrical Load	No. Units	Duty Cycle	CB Amps	Max. Drain Amps (28 Vdc)
TIE	Master Contactor	Relay	1	Cont.	N/A	.60
TIE	Volt/Amp/Pwr Supply		1	Cont.		1.20
	Volt / Ammeter Solid State	Meter Inductive	1/2	Cont.	N/A	N/A
M	Vent / Defog Blower	Blower Motor	1	Int.	10	7
		Relay	1			.35
M	Stall Warn. Heat	Heating Element	2	Int.	25	16.10
		Relay	2	Int.		.35
M	Heated Windshield	Heating Elements	2	Int.	25	HI 20.74/ LO 13.59
M		Relay	2	Int.		
M		Controller and Relay	1	Int.	5	1.40
M	Pitot Heat	Heating Elements	1 (2)	Cont.	10	3.2 (6)
		Relay	1	Int.		.07
	Propeller Heat					
M	3-Blade	Heating Elements	3	Int.	20	18
	2-Blade		2	Int.	20	10
		Timer	1	Int.		
		Relay	1	Int.		.29
M	Surface Deice	Timer	1	Int.	5	1.85
M	Alternator Field 1/2	Voltage Regulator Solid State	2	Cont.	3.0	1.50
M	Starter	Contactor	1	Int.	5.0	1.47
		Motor	1	Int.		185.0
M	Annunciator Panel	Lamps	1	Int.	3	.46
M	Fuel Pump	Motor	2	Int.	5	4.0
M	Hydraulic Pump	Motor	1	Int.	25	26.0
M		Relay	1		.35	.35
		Control	1		3	
M	Strobe Light	Light	2/3	Int.	5	4.0
M	Landing Light	Lamp - (250W)	1	Int.	10	8.9
		Relay			.29	.07
M	Taxi Lights	Lamps - (50W)	2	Int.	5	3.5
		Relay	1			
M	Position Lights	Lamps	2	Cont.	5	3.0
M	Wing Inspect. (Ice) Light	Lamp	1	Int.	3	0.9
	Map Lights	Lamps	2	Int.	3	.60
M	Instrument Lights	Lamps	28	Cont.	5	2.00
M	Gear Position Lights	Lamps	3	Cont.	3	.24
M	Stall and Gear Warning Horn	Horn	1	Int.	3	.25
M	Avionics Cooling	Blower Motor	1	Cont.	3	0.50
M	Eng. Gauge Cluster (1)	Oil Pressure Oil Temperature Cylinder Head Temperature	1	Cont.	3	.30
M	Cylinder Head Scanner	All Cylinder Head Temperatures	1	Cont.	1.5	.25
M	CHT/VAC Ind.			Cont.		0.25
M	MAP/RPM Ind.			Cont.		0.25
M	Oil Press/Oil Temp Ind.			Cont.		0.25
M	TIT/ FF Ind.			Cont.		0.25
M	E. L. Panels Inverter			Cont.		0.50
M	Electric Flaps			Int.		6.00

CHART 1 (Sheet 2 of 3)
ELECTRICAL LOAD ANALYSIS (PA-46-350P 1995–2010)

Bus	Component / System	Electrical Load	No. Units	Duty Cycle	CB Amps	Max. Drain Amps (28 Vdc)
M	Emergency Fuel Pump			Int.		4.50
M	Fuel Quantity (L & R)			Cont.		0.25
M	Ignition			Int.		3.50
M	Regulator Alt. #1			Cont.		0.07
M	Regulator Alt. #2			Cont.		0.07
M	Turn Coordinator(s)		1 (2)	Cont.	3	0.30 (0.60)
M	MFD		1	Cont.	7.5	2.10
M	PFD		2	Cont.	10	3
M	S-TEC Autopilot			Cont.	5	0.90
NE	Cabin Heat	Heater	1	Int.	35 Fuse	33.0
NE		Relay	1			.35
NE	Cabin Lights	Lamps	4	Int.	5	.30
NE	Air Conditioning	Condenser Blower	1	Int.	25	16.50
NE		Cabin Fans	2	Int.	15	12.0
NE		Control, Clutch, and Relay	1	Int.	5	2.29
NE	Avionics Lighting			Cont.	3	1.00
NE	Cabin Dump Solenoid			Int.		0.29
NE	CD Player			Cont.		0.20
NE	Hourmeter			Cont.		0.05
NE	Ice Detector			Cont.		1.00
NE	Inverter			Int.		2.70
AVI	Transicoil EMIS	Quad Digital Indicator (2)	1	Cont.	3	.14
	Digital Indicators	Enhanced Digital Indicator (3)	1	Cont.	3	.14
AVI	Transicoil EMIS	Oil Pressure/Oil Temperature	1	Cont.	5	.50
	Analog Indicators (4)	MAP	1	Cont.	3	.25
		RPM	1	Cont.	3	.25
		TIT	1	Cont.	3	.25
		Fuel Flow	1	Cont.	3	.25
		CHT	1	Cont.	3	.25
		Fuel Quantity	1	Cont.	5	.50
AVI	#1 KX-155A COMM/NAV			Cont.		0.06/6.0
AVI	#2 KX-155A COMM/NAV			Cont.		0.06/6.0
AVI	Argus 7000			Cont.		0.54
AVI	Avionics #1 Contactor			Int.		0.70
AVI	Avionics #2 Contactor			Int.		0.70
AVI	Clock DVR 3001-XT			Cont.		0.10
AVI	GMA-430 Audio Panel		1	Cont.	5	0.50
AVI	GNS-430 COM/NAV/GPS #1		1	Cont.	5	1.50
AVI	GNS-430 COM/NAV/GPS #2		1	Cont.	5	1.50
AVI	GTX-330 Transponder		2	Cont.	5	1.00
AVI	KAS-297 Alt. Alert			Cont.		0.70
AVI	KCS-55A Compass			Cont.		1.60
AVI	KEA-130A Encoding Alt.			Cont.		0.24
AVI	KEA-346 Encoding Alt.			Cont.		0.24
AVI	KFC-150 Autopilot			Int.		8.73
AVI	KLN-90B GPS			Cont.		1.00
AVI	KMH-820 IHAS			Cont.	3	1.65
AVI	KN-40			Cont.		1.00

CHART 1 (Sheet 3 of 3)
ELECTRICAL LOAD ANALYSIS (PA-46-350P 1995–2010)

Bus	Component / System	Electrical Load	No. Units	Duty Cycle	CB Amps	Max. Drain Amps (28 Vdc)
AVI	KN-63 DME			Cont.	3	0.60
AVI	KN-72 Converter			Cont.		0.05
AVI	KR-87 ADF			Cont.	2	0.45
AVI	KRA-10A Radar Altimeter			Cont.		0.20
AVI	KT-76C Transponder			Cont.		0.70
AVI	PMA 6000 M-S			Cont.		1.00
AVI	RDR-2000 Radar			Cont.		4.30
AVI	SG-465			Cont.		3.20
AVI	Vibrator			Int.		2.00
AVI	WX-500 Stormscope			Cont.	3	0.38
AVI	WX-1000 Stormscope			Cont.		1.00
AVI	XM Satellite Receiver			Cont.	1	0.20
AVI	Yaw Damper			Int.		1.80

NOTES:
(1) S/N's 4636001 thru 4636020 only.
(2) S/N's 4636021 thru 4636131 only.
(3) S/N's 4636132 thru 4636374.
(4) S/N's 4636021 thru 4636374.

General: "Bus" Abbreviations: AVI = Avionics, M = Main, NE = Non-Essential, TIE = Tie
Avionics listed above include optional installations.
"# Units" default value is one.
"Duty Cycle" = Intermittent or Continuous.
Annunciator Control bulbs (.04 EA.) are normally OFF.
Windshield Heat Relays and High and Lo are each 50% duty cycle.
Prop Feather engages at engine shutdown.
Rudder Trim is 20% duty cycle.
S-TEC Autopilot draws 0.68 for 50% and 0.90 for 50% when in cruise; and 0.68 during takeoff and landing.

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633

CHART 2 (Sheet 1 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46-350P MIRAGE G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
AIRCRAFT ELECTRIC LOADS														
Battery Master	N/A	0.6	CONT	1	15	15	60	10		0.6	9	9	36	6
Starter	N/A	110	INT	0.5	0	0	0	0	Three 10-sec engine crank cycles	55	0	0	0	0
Starter Contactor	309	1.47	INT	0.5	0	0	0	0	Same as engine starter	0.735	0	0	0	0
Starter Vibrator	309	2	INT	0.5	0	0	0	0	Same as engine starter	1	0	0	0	0
Alt #1 Field & Regulator	310	1.57	CONT	0	15	15	60	10	1.5 field & 0.07 regulator	0	23.55	23.55	94.2	15.7
Alt #2 Field & Regulator	311	1.57	CONT	0	15	15	60	10	1.5 field & 0.07 regulator	0	23.55	23.55	94.2	15.7
Avionics #1 Contatctor	307	0.7	INT	1	0	0	0	0	Avionics ON = de-energized	0.7	0	0	0	0
Avionics #2 Contatctor	308	0.7	INT	1	0	0	0	10	Avionics ON = de-energized	0.7	0	0	0	0
Engine Hour Meter	340	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
L/R Fuel Boost Pump	318 & 319	4	CONT	1	15	15	60	10	One pump in continous operation	4	60	60	240	40
Emergency Fuel Pump	317	4.5	INT	1	0	3	0	5	Start T/O, Landing & Emergency only	4.5	0	13.5	0	22.5
Landing Gear Pos. Indicator	327	0.24	CONT	1	15	2	0	5		0.24	3.6	0.48	0	1.2
Gear Unsafe Annunciator	328	0.08	INT	0	0	0	0	0.2	(12 seconds)	0	0	0	0	0.16
Hydraulic Pump	326	26	INT	0	0	0.1	0	0.1	During Gear Ext. & Retraction only	0	0	2.6	0	2.6
Hydraulic Pump Relay	325	0.35	INT	0	0	0.1	0	0.1	During Gear Ext & Retraction only	0	0	0.035	0	0.035
Gear/Stall Warn Horn	328	0.25	INT	0	0	0	0	0	Negligible	0	0	0	0	0

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633

CHART 2 (Sheet 2 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46-350P MIRAGE G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
Stall Warning Computer	329	0.3	CONT	1	15	15	60	10	Safe Flight Comp.	0.3	4.5	4.5	18	3
Stall Warning Heat	342	16.1	CONT	0	15	15	60	10		0	241.5	241.5	966	161
Speed Brakes	330	0.44	CONT	1	15	15	0	10	2.15 amps intermittent	0.44	6.6	6.6	26.4	4.4
Flap Motor	331	6	INT	0	0.1	0	0	0.1	T/O @ 10 Deg, Landing @ 36 Deg.	0	0.6	0	0	0.6
Flap Warning	332	0.15	INT	0	0.1	0	60	0.1	Only during transit	0	0.015	0	0	0.015
Left Pitot Heat	343	6	CONT	0	15	15	60	10	50% on ground	0	45	90	360	60
Right Pitot Heat	344	6	CONT	0	15	15	0	10	50% on ground	0	45	90	360	60
Annunciator Lights	320	0.24	INT	0	0	0	60	0	Bulbs @0.04 each, normally OFF	0	0	0	0	0
Anti-collision Lights	315	4.7	CONT	0	0	15	60	10		0	0	70.5	282	47
Position (NAV) Lights	314	3.8	CONT	0	15	15	0	10		0	57	57	228	38
Taxi Lights (2@3.5)	313	7	CONT	0	15	0	0	0		0	105	0	0	0
Landing Light	312	8.9	CONT	0	0	5	0	5		0	0	44.5	0	44.5
Landing Light Relay	312	0.7	CONT	0	0	5	0	5	Same as Landing Light	0	0	3.5	0	3.5
Avionics Lights	321	1	CONT	0	15	15	60	10	@ 50% intensity	0	7.5	15	60	10
I.P. Lights /E.L. Placards	322	1.7	CONT	0	15	15	0	10	Includes switch lts	0	25.5	25.5	102	17
Reading (cabin) Lights	323	10	INT	0	1	0	0	0	Normally off during flight (5 @ 2 amps)	0	10	0	0	0
Cockpit Map Light	324	2	INT	0	5	0	0	0	Pre-take off cklst	0	10	0	0	0
Vent Defog Blower	341	7	CONT	0	15	7	0	7		0	105	49	0	49
Cabin Fresh Air Blower (2 @ 6 ea)	339	12	CONT	0	7.5	7.5	0	0	Not required in flight	0	90	90	0	0
Cabin Press. Dump Sol.	353	0.29	CONT	1	15	0	00	0		0.29	4.35	0	0	0

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633

CHART 2 (Sheet 3 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46-350P MIRAGE G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	Taxi	T/O	Cruise	Land
Vacuum	354	0.025	CONT	0	15	15	60	10	Vac. Transducer only	0	0.375	0.375	1.5	0.25
Left Fuel Level	355 N/A	--	N/A	0	--	--	--	--	Reserved for future	--	--	--	--	--
Right Fuel Level	356 N/A	--	N/A	0	--	--	--	--	Reserved for future	--	--	--	--	--
A/C Control	357	2	CONT	0	15	7.5	0	0	Clutch & Relay	0	30	15	0	0
A/C Power	338	16.5	CONT	0	15	7.5	0	0	Condenser blower	0	247.5	123.75	0	0
Prop De-Ice	345	18	CONT	0	15	15	60	10	50% duty cycle	0	135	135	540	90
Surface De-Ice	346	1.85	CONT	0	0	15	60	10	50% duty cycle	0	0	13.875	55.5	9.25
Windshield Heat Control	347	1.5	CONT	0	15	15	60	10	50% duty cycle	0	11.25	11.25	45	7.5
Windshield Heat (Low)	348	13.59	CONT	0	15	0	0	0	50% duty cycle	0	101.925	0	0	0
Windshield Heat (High)	348	20.74	CONT	0	0	15	60	10	50% duty cycle	0	0	155.55	622.2	103.7
Wing Ice Light	316	0.9	INT	0	0.5	0	1	1	Occasional intermittent use	0	0.45	0	0.9	0.9
Cabin Heater	N/A	33	INT	0	15	0	0	0	35 amp fuse off battery feed. Used only during taxi	0	495	0	0	0
Cabin Heater Relay	N/A	0.35	INT	0	15	0	0	0		0	5.25	0	0	0
Inverter (110-V AC Power)	336	12	CONT	0	0	7.5	60	0	50% load max in cruise	0	0	90	72	0
Lightning Protection	349/387	--	N/A	0	0	0	0	0	N/A only active during a lightning attachment event	0	0	0	0	0
AIRCRAFT AVIONICS LOADS														
GDU 1040 #1 PFD	359	1.4	CONT	1	15	15	60	10		1.4	21	21	84	14
PFD #1 Fan	358	0.4	CONT	1	15	15	60	10		0.4	6	6	24	4
GDU 1040 #2 PFD	389	1.4	CONT	0	15	15	60	10		0	21	21	84	14

246010

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633

CHART 2 (Sheet 4 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46-350P MIRAGE G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	Taxi	T/O	Cruise	Land
PFD #2 Fan	390	0.4	CONT	0	15	15	60	10		0	6	6	24	4
GDU 1500 MFD	375	2.3	CONT	1	15	15	60	10		2.3	34.5	34.5	138	23
MFD & Avionics Fans	376	0.8	CONT	1	15	15	60	10	Two fans	0.8	12	12	48	8
GDL 69A Data Link	379	0.35	CONT	0	15	15	60	10		0	5.25	5.25	21	3.5
GRT 10 Transceiver	379	0.044	CONT	0	15	15	60	10		0	0.66	0.66	2.64	0.44
GRC 10 Remote Control	N/A	0	N/A	0	15	15	60	10		0	0	0	0	0
GWX 68 WX Radar	373	2	CONT	0	5	15	60	5		0	10	30	120	10
GMC 710 A/P Mode Control	378	0.183	CONT	0	15	15	60	10		0	2.745	2.745	10.98	1.83
GEA 71 Eng/Airframe	357	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
GCU 476 Keypad	377	0.16	CONT	1	15	15	60	10		0.16	2.4	2.4	9.6	1.6
GIA 63W #1 IAU (Tx)	363	4	INT	0	2	1	12	3	Increased current for transmit only	0	8	4	48	12
GIA 63W #1 IAU (Rx)	364	1.3	CONT	1	15	15	60	10		1.3	19.5	19.5	7.8	13
GIA 63W #2 IAU (Rx)	383	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
GIA 63W #2 IAU (Rx)	382	1.3	CONT	0	15	15	60	10		0	22.425	22.425	89.7	14.96
GDC 74A #1 ADC	361	0.23	CONT	1	15	15	60	10		0.23	3.45	3.45	13.8	2.3
GDC 74A #2 ADC	385	0.23	CONT	0	15	15	60	10		0	3.45	3.45	13.8	2.3
GTP 59 #1 OAT Probe	N/A	--	N/A	1	15	15	60	10	Included with power to ADC	0	0	0	0	0
GTP 59 #2 OAT Probe	N/A	--	N/A	0	15	15	60	10		0	0	0	0	0
GRS 77 #1 AHRS	362	0.344	CONT	1	15	15	60	10		0.344	5.16	5.16	20.64	3.44
GRS 77 #2 AHRS	384	0.344	CONT	0	15	15	60	10		0	5.16	5.16	20.64	3.44
GMU 44 #1Magnetometer	N/A	0.3	N/A	1	15	15	60	10	Included with power to AHRS	0.3	4.5	4.5	18	3
GMU 44 #2Magnetometer	N/A	0.3	NA/	0	15	15	60	10		0	4.5	4.5	18	3
GMA 347 Audio Panel & Marker Beacon	360	1.95	CONT	1	15	15	60	10		1.95	29.55	29.55	11	19.5
GTX 33 #1 Transponder	374	1.83	CONT	0	15	15	60	10	Transponder 1 or 2 in operation	0	27.45	27.45	109.8	18.3

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633

CHART 2 (Sheet 5 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46-350P MIRAGE G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	Taxi	T/O	Cruise	Land
GTX 33 #2 Transponder	395	0.85	STBY	0	15	15	60	10	Transponder 1 or 2 in operation	0	12.75	12.75	51	8.5
GAE 43 Alt Data System		0.1	CONT	0	15	15	60	10		0	1.5	1.5	6	1
Pitch Trim (Manual)	333	1.6	INT	0	0	0	0	0	Included with A/P calculation below	0	0	0	0	0
A/P with 4 Servos	334	8.19	CONT	0	7.5	15	60	10	Estimated 50% duty cycle	0	9.21	18.43	73.71	12.285
Standby Attitude Ind.	335	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
KR-87 ADF	394	0.635	CONT	0	15	15	60	10		0	9.525	9.525	38.1	6.35
KN-63 DME	393	1.5	CONT	0	15	15	60	10		0	22.5	22.5	90	15
WX-500 StormScope	391	0.38	CONT	0	15	15	60	10		0	5.7	5.7	22.8	3.8
KTA-810 TAS	392	1.65	CONT	0	15	15	60	10		0	24.75	24.75	99	16.5
									TOTAL AMP MINUTES	78.45	2194.70	1781.02	6147.51	1023.3
									UTILIZATION LOAD (AMPS)		146.31	118.73	102.46	102.33

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Effectivity
4692134 and up, less 4692141
4692149 and 4692153

CHART 3 (Sheet 1 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46R-350T MATRIX G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	Taxi	T/O	Cruise	Land
AIRCRAFT ELECTRIC LOADS														
Battery Master	N/A	0.6	CONT	1	15	15	60	10		0.6	9	9	36	6
Starter	N/A	110	INT	0.5	0	0	0	0	Three 10-sec engine crank cycles	55	0	0	0	0
Starter Contactor	309	1.47	INT	0.5	0	0	0	0	Same as engine starter	0.735	0	0	0	0
Starter Vibrator	309	2	INT	0.5	0	0	0	0	Same as engine starter	1	0	0	0	0
Alt #1 Field & Regulator	310	1.57	CONT	0	15	15	60	10	1.5 field & 0.07 regulator	0	23.55	23.55	94.2	15.7
Alt #2 Field & Regulator	311	1.57	CONT	0	15	15	60	10	1.5 field & 0.07 regulator	0	23.55	23.55	94.2	15.7
Avionics #1 Contatctor	307	0.7	INT	1	0	0	0	0	Avionics ON = de-energized	0.7	0	0	0	0
Avionics #2 Contatctor	308	0.7	INT	1	0	0	0	10	Avionics ON = de-energized	0.7	0	0	0	0
Engine Hour Meter	340	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
L/R Fuel Boost Pump	318 & 319	4	CONT	1	15	15	60	10	One pump in continous operation	4	60	60	240	40
Emergency Fuel Pump	317	4.5	INT	1	0	3	0	5	Start T/O, Landing & Emergency only	4.5	0	13.5	0	22.5
Landing Gear Pos. Indicator	327	0.24	CONT	1	15	2	0	5		0.24	3.6	0.48	0	1.2
Gear Unsafe Annunciator	328	0.08	INT	0	0	0	0	0.2	(12 seconds)	0	0	0	0	0.16
Hydraulic Pump	326	26	INT	0	0	0.1	0	0.1	During Gear Ext. & Retraction only	0	0	2.6	0	2.6
Hydraulic Pump Relay	325	0.35	INT	0	0	0.1	0	0.1	During Gear Ext & Retraction only	0	0	0.035	0	0.035
Gear/Stall Warn Horn	328	0.25	INT	0	0	0	0	0	Negligible	0	0	0	0	0

Effectivity
4692134 and up, less 4692141
4692149 and 4692153

CHART 3 (Sheet 2 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46R-350T MATRIX G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
Stall Warning Computer	329	0.3	CONT	1	15	15	60	10	Safe Flight Comp.	0.3	4.5	4.5	18	3
Stall Warning Heat	342	16.1	CONT	0	15	15	60	10		0	241.5	241.5	966	161
Speed Brakes	330	0.44	CONT	1	15	15	0	10	2.15 amps intermittent	0.44	6.6	6.6	26.4	4.4
Flap Motor	331	6	INT	0	0.1	0	0	0.1	T/O @ 10 Deg, Landing @ 36 Deg.	0	0.6	0	0	0.6
Flap Warning	332	0.15	INT	0	0.1	0	60	0.1	Only during transit	0	0.015	0	0	0.015
Left Pitot Heat	343	6	CONT	0	15	15	60	10	50% on ground	0	45	90	360	60
Right Pitot Heat	344	6	CONT	0	15	15	0	10	50% on ground	0	45	90	360	60
Annunciator Lights	320	0.24	INT	0	0	0	60	0	Bulbs @0.04 each, normally OFF	0	0	0	0	0
Anti-collision Lights	315	4.7	CONT	0	0	15	60	10		0	0	70.5	282	47
Position (NAV) Lights	314	3.8	CONT	0	15	15	0	10		0	57	57	228	38
Taxi Lights (2@3.5)	313	7	CONT	0	15	0	0	0		0	105	0	0	0
Landing Light	312	8.9	CONT	0	0	5	0	5		0	0	44.5	0	44.5
Landing Light Relay	312	0.7	CONT	0	0	5	0	5	Same as Landing Light	0	0	3.5	0	3.5
Avionics Lights	321	1	CONT	0	15	15	60	10	@ 50% intensity	0	7.5	15	60	10
I.P. Lights /E.L. Placards	322	1.7	CONT	0	15	15	0	10	Includes switch lts	0	25.5	25.5	102	17
Reading (cabin) Lights	323	10	INT	0	1	0	0	0	Normally off during flight (5 @ 2 amps)	0	10	0	0	0
Cockpit Map Light	324	2	INT	0	5	0	0	0	Pre-take off cklst	0	10	0	0	0
Vent Defog Blower	341	7	CONT	0	15	7	0	7		0	105	49	0	49
Cabin Fresh Air Blower (2 @ 6 ea)	339	12	CONT	0	7.5	7.5	0	0	Not required in flight	0	90	90	0	0
Cabin Press. Dump (N/A)	353	0	N/A	-	-	-	-	-		-	-	-	-	-

Effectivity
4692134 and up, less 4692141
4692149 and 4692153

CHART 3 (Sheet 3 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46R-350T MATRIX G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	Taxi	T/O	Cruise	Land
Vacuum	354	0.025	CONT	0	15	15	60	10	Vac. Transducer only	0	0.375	0.375	1.5	0.25
Left Fuel Level	355 N/A	--	N/A	0	--	--	--	--	Reserved for future	--	--	--	--	--
Right Fuel Level	356 N/A	--	N/A	0	--	--	--	--	Reserved for future	--	--	--	--	--
A/C Control	357	2	CONT	0	15	7.5	0	0	Clutch & Relay	0	30	15	0	0
A/C Power	338	16.5	CONT	0	15	7.5	0	0	Condenser blower	0	247.5	123.75	0	0
Prop De-Ice	345	18	CONT	0	15	15	60	10	50% duty cycle	0	135	135	540	90
Surface De-Ice	346	1.85	CONT	0	0	15	60	10	50% duty cycle	0	0	13.875	55.5	9.25
Windshield Heat Control	347	1.5	CONT	0	15	15	60	10	50% duty cycle	0	11.25	11.25	45	7.5
Windshield Heat (Low)	348	13.59	CONT	0	15	0	0	0	50% duty cycle	0	101.925	0	0	0
Windshield Heat (High)	348	20.74	CONT	0	0	15	60	10	50% duty cycle	0	0	155.55	622.2	103.7
Wing Ice Light	316	0.9	INT	0	0.5	0	1	1	Occasional intermittent use	0	0.45	0	0.9	0.9
Cabin Heater	N/A	33	INT	0	15	0	0	0	35 amp fuse off battery feed. Used only during taxi	0	495	0	0	0
Cabin Heater Relay	N/A	0.35	INT	0	15	0	0	0		0	5.25	0	0	0
Inverter (110-V AC Power)	336	12	CONT	0	0	7.5	60	0	50% load max in cruise	0	0	90	72	0
Lightning Protection	349/387	--	N/A	0	0	0	0	0	N/A only active during a lightning attachment event	0	0	0	0	0
AIRCRAFT AVIONICS LOADS														
GDU 1040 #1 PFD	359	1.4	CONT	1	15	15	60	10		1.4	21	21	84	14
PFD #1 Fan	358	0.4	CONT	1	15	15	60	10		0.4	6	6	24	4
GDU 1040 #2 PFD	389	1.4	CONT	0	15	15	60	10		0	21	21	84	14

Effectivity
4692134 and up, less 4692141
4692149 and 4692153

CHART 3 (Sheet 4 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46R-350T MATRIX G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
PFD #2 Fan	390	0.4	CONT	0	15	15	60	10		0	6	6	24	4
GDU 1500 MFD	375	2.3	CONT	1	15	15	60	10		2.3	34.5	34.5	138	23
MFD & Avionics Fans	376	0.8	CONT	1	15	15	60	10	Two fans	0.8	12	12	48	8
GDL 69A Data Link	379	0.35	CONT	0	15	15	60	10		0	5.25	5.25	21	3.5
GRT 10 Transceiver	379	0.044	CONT	0	15	15	60	10		0	0.66	0.66	2.64	0.44
GRC 10 Remote Control	N/A	0	N/A	0	15	15	60	10		0	0	0	0	0
GWX 68 WX Radar	373	2	CONT	0	5	15	60	5		0	10	30	120	10
GMC 710 A/P Mode Control	378	0.183	CONT	0	15	15	60	10		0	2.745	2.745	10.98	1.83
GEA 71 Eng/Airframe	357	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
GCU 476 Keypad	377	0.16	CONT	1	15	15	60	10		0.16	2.4	2.4	9.6	1.6
GIA 63W #1 IAU (Tx)	363	4	INT	0	2	1	12	3	Increased current for transmit only	0	8	4	48	12
GIA 63W #1 IAU (Rx)	364	1.3	CONT	1	15	15	60	10		1.3	19.5	19.5	7.8	13
GIA 63W #2 IAU (Rx)	383	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
GIA 63W #2 IAU (Rx)	382	1.3	CONT	0	15	15	60	10		0	22.425	22.425	89.7	14.96
GDC 74A #1 ADC	361	0.23	CONT	1	15	15	60	10		0.23	3.45	3.45	13.8	2.3
GDC 74A #2 ADC	385	0.23	CONT	0	15	15	60	10		0	3.45	3.45	13.8	2.3
GTP 59 #1 OAT Probe	N/A	--	N/A	1	15	15	60	10	Included with power to ADC	0	0	0	0	0
GTP 59 #2 OAT Probe	N/A	--	N/A	0	15	15	60	10		0	0	0	0	0
GRS 77 #1 AHRS	362	0.344	CONT	1	15	15	60	10		0.344	5.16	5.16	20.64	3.44
GRS 77 #2 AHRS	384	0.344	CONT	0	15	15	60	10		0	5.16	5.16	20.64	3.44
GMU 44 #1Magnetometer	N/A	0.3	N/A	1	15	15	60	10	Included with power to AHRS	0.3	4.5	4.5	18	3
GMU 44 #2Magnetometer	N/A	0.3	NA/	0	15	15	60	10		0	4.5	4.5	18	3
GMA 347 Audio Panel & Marker Beacon	360	1.95	CONT	1	15	15	60	10		1.95	29.55	29.55	11	19.5
GTX 33 #1 Transponder	374	1.83	CONT	0	15	15	60	10	Transponder 1 or 2 in operation	0	27.45	27.45	109.8	18.3

Effectivity
4692134 and up, less 4692141
4692149 and 4692153

CHART 3 (Sheet 5 of 5)
ELECTRICAL LOAD ANALYSIS (PA-46R-350T MATRIX G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
GTX 33 #2 Transponder	395	0.85	STBY	0	15	15	60	10	Transponder 1 or 2 in operation	0	12.75	12.75	51	8.5
GAE 43 Alt Data System		0.1	CONT	0	15	15	60	10		0	1.5	1.5	6	1
Pitch Trim (Manual)	333	1.6	INT	0	0	0	0	0	Included with A/P calculation below	0	0	0	0	0
A/P with 4 Servos	334	8.19	CONT	0	7.5	15	60	10	Estimated 50% duty cycle	0	9.21	18.43	73.71	12.285
Standby Attitude Ind.	335	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
KR-87 ADF	394	0.635	CONT	0	15	15	60	10		0	9.525	9.525	38.1	6.35
KN-63 DME	393	1.5	CONT	0	15	15	60	10		0	22.5	22.5	90	15
WX-500 StormScope	391	0.38	CONT	0	15	15	60	10		0	5.7	5.7	22.8	3.8
KTA-810 TAS	392	1.65	CONT	0	15	15	60	10		0	24.75	24.75	99	16.5
									TOTAL AMP MINUTES	78.45	2190.35	1781.02	6147.51	1023.3
									UTILIZATION LOAD (AMPS)		146.02	118.73	102.46	102.33

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Effectivity
4636633, 4636652 and up

CHART 4 (Sheet 1 of 6)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
AIRCRAFT ELECTRIC LOADS														
BATTERY MASTER RELAY	BATT/ EXT	0.6	CONT	1	15	15	60	10		0.6	9	9	36	6
STARTER	BATT/ EXT	110	INT	0.5	0	0	0	0	THREE 10 SEC ENGINE CRANK CYCLES	55	0	0	0	0
STARTER CONTACTOR	309	1.47	INT	0.5	0	0	0	0	SAME AS ENGINE STARTER	0.735	0	0	0	0
STARTER VIBRATOR	309	2	INT	0.5	0	0	0	0	SAME AS ENGINE STARTER	1	0	0	0	0
ALT #1 FIELD & REGULATOR	310	1.57	CONT	0	15	15	60	10	1.5 FIELD AND .07 REGULATOR CURRENT	0	23.55	23.55	94.2	15.7
ALT #2 FIELD & REGULATOR	311	1.57	CONT	0	15	15	60	10	1.5 FIELD AND .07 REGULATOR CURRENT	0	23.55	23.55	94.2	15.7
AVIONICS #1 CONTACTOR	307	0.7	INT	1	0	0	0	0	AVIONICS ON = DEENERGIZED	0.7	0	0	0	0
AVIONICS #2 CONTACTOR	308	0.7	INT	1	0	0	0	0	AVIONICS ON = DEENERGIZED	0.7	0	0	0	0
ENGINE HOUR METER	340	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
LEFT/ RIGHT FUEL BOOST PUMP	318 / 319	4	CONT	1	15	15	60	10	ONE PUMP IN CONTINUOUS OPERATION	4	60	60	240	40
EMERGENCY FUEL PUMP	317	4.5	INT	1	0	3	0	5	START, T/O, LANDING & EMERGENCY	4.5	0	13.5	0	22.5
HYDRAULIC PUMP	326	26	INT	0	0	0.1	0	0.1	DURING GEAR EXT/RET ONLY (6 - 7 SECONDS)	0	0	2.6	0	2.6

Effectivity
4636633, 4636652 and up

CHART 4 (Sheet 2 of 6)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
HYDRAULIC PUMP CONTROL RELAY	325	0.35	INT	0	0	0.1	0	0.1	DURING GEAR EXT/RET ONLY (6 - 7 SECONDS)	0	0	0.035	0	0.035
STALL WARNING (COMPUTER)	327	0.3	CONT	1	15	15	60	10	SAFE FLIGHT COMPUTER	0.3	4.5	4.5	18	3
STALL WARNING HEAT	342	16.1	CONT	0	15	15	60	10		0	241.5	241.5	966	161
SPEED BRAKES	328	0.44	CONT	1	15	15	60	10	2.15 AMPS INTERMITTENT AT DEPLOY	0.44	6.6	6.6	26.4	4.4
ELECTRIC FLAPS	331	6	INT	0	0.1	0	0	0.1	T/O @ 10 DEG. / LDG AT 36 DEG	0	0.6	0	0	0.6
FLAP WARNING	332	0.15	INT	0	0.1	0	0	0.1	ONLY DURING TRANSIT	0	0.015	0	0	0.015
LEFT PITOT HEAT	343	6	CONT	0	15	15	60	10	50% ON GND / 100% IN FLIGHT	0	45	90	360	60
RIGHT PITOT HEAT	344	6	CONT	0	15	15	60	10	50% ON GND / 100% IN FLIGHT	0	45	90	360	60
ANTI-COLLISION STROBES - (2 @ .5 EACH)	315	1	CONT	0	0	15	60	10		0	0	15	60	10
POSITION (NAV) LIGHTS - LED (2 @ 0.17 EACH)	314	0.34	CONT	0	15	15	60	10		0	5.1	5.1	20.4	3.4
TAXI LED LIGHT (2 @ 0.45 EACH)	313	0.9	CONT	0	15	0	0	0		0	13.5	0	0	0
LANDING LED LIGHT	312	0.6	CONT	0	0	5	0	5		0	0	3	0	3
LANDING LIGHTS RELAY	312	0.7	CONT	0	0	5	0	5	SAME AS LANDING LIGHT	0	0	3.5	0	3.5
AVIONICS LIGHTS (DIMMER)	321	1	CONT	0	15	15	60	10	@ 50% INTENSITY	0	7.5	15	60	10
INSTRUMENT PANEL LIGHTS / E.L. PLACARDS	322	1.7	CONT	0	15	15	60	10	INCLUDES SWITCH LIGHTS (@50% INTENSITY)	0	25.5	25.5	102	17

Effectivity
4636633, 4636652 and up

CHART 4 (Sheet 3 of 6)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	AMP-MINUTES Taxi	AMP-MINUTES T/O	AMP-MINUTES Cruise	AMP-MINUTES Land
READING LIGHTS (CABIN)	323	0.082 (8 ea.)	INT	0	1	0	0	0	NORMALLY OFF DURING FLIGHT (5 @ 2 AMPS EA)	0	10	0	0	0
COCKPIT FLOOD LIGHT (MAP)	324	0.082 (2 ea.)	INT	0	5	0	0	0	PRE-TAKE OFF CHECKLIST	0	10	0	0	0
VENT DEFOG BLOWER	341	7	CONT	0	15	7	0	7		0	105	49	0	49
CABIN FRESH AIR BLOWER (2 @ 6 EA)	339	12	CONT	0	7.5	7.5	0	0	NOT REQUIRED IN FLIGHT	0	90	90	0	0
CPCS CTRL	352	0.47	CONT	1	15	15	60	10		0.47	7.05	7.05	28.2	4.7
CABIN PRESSURE DUMP SOLENOID	353	0.29	CONT	1	15	0	0	0		0.29	4.35	0	0	0
VACUUM	354	0.025		0	15	15	60	10	VACUUM TRANSDUCER ONLY (<25 ma load)	0.00	0.38	0.38	1.50	0.25
LH/RH Ametek Capacitance Fuel Quantity	357	0.07		0	15	15	60	10		0	1.05	1.05	4.2	0.7
AIR CONDITIONING CONTROL *	337	2		0	0	0	0	0	CLUTCH & RELAY; TAXI & CLIMBOUT	0	0	0	0	0
AIR CONDITIONING POWER *	338	16.5		0	0	0	0	0	EVAPORATER BLOWER; TAXI & CLIMBOUT	0	0	0	0	0
PROPELLER DE-ICE	345	18	CONT	0	15	15	60	10	50% DUTY CYCLE	0	135	135	540	90
SURFACE DE-ICE	346	1.85	CONT	0	0	15	60	10	50% DUTY CYCLE	0	0	13.875	55.5	9.25
WINDSHIELD HEAT RELAYS (CONTROL)	347	1.5	CONT	0	15	15	60	10	50% DUTY CYCLE	0	11.25	11.25	45	7.5
WINDSHIELD HEAT LOW	348	13.59	CONT	0	15	0	0	0	50% DUTY CYCLE	0	101.925	0	0	0
WINDSHIELD HEAT HIGH	348	20.74	CONT	0	0	15	60	10	50% DUTY CYCLE	0	0	155.55	622.2	103.7

Effectivity
4636633, 4636652 and up

CHART 4 (Sheet 4 of 6)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	Taxi	T/O	Cruise	Land
WING ICE LIGHT	316	0.9	INT	0	0.5	0	1	1	OCCASIONAL INTERMITTENT USE IN CRUISE	0	0.45	0	0.9	0.9
CABIN HEATER	301	33	INT	0	15	0	0	0	35 AMP FUSE OFF BATTERY FEED; USED ONLY	0	495	0	0	0
CABIN HEATER RELAY	301	0.35	INT	0	15	0	0	0	DURING TAXI, OTHERWISE OFF.	0	5.25	0	0	0
INVERTER - 110 VOLT AC POWER	336	12	CONT	0	0	7.5	60	0	50% LOADED MAXIMUM IN CRUISE	0	0	90	720	0
LIGHTNING PROTECTION	349 / 380	N/A	INT	0	0	0	0	0	N/A - ONLY DURING LIGHTNING EVENT	0	0	0	0	0
AIRCRAFT AVIONICS LOADS														
GDU 1040 #1 PFD	359	1.4	CONT	1	15	15	60	10		1.4	21	21	84	14
PFD #1 FAN	358	0.4	CONT	1	15	15	60	10		0.4	6	6	24	4
GDU 1040 #2 PFD	389	1.4	CONT	0	15	15	60	10		0	21	21	84	14
PFD #2 FAN	390	0.4	CONT	0	15	15	60	10		0	6	6	24	4
GDU 1240A MFD	375	2.7	CONT	1	15	15	60	10		2.7	40.5	40.5	162	27
MFD FAN & AVIONICS FAN	376	0.8	CONT	1	15	15	60	10		0.8	12	12	48	8
AFT Avionics Cooling	376	0.4	CONT	0	15	15	60	10		0	6	6	24	4
GDL WX DATALINK	379	0.35	CONT	0	15	15	60	10		0	5.25	5.25	21	3.5
GRT 10 DATALINK TRANSCEIVER	379	0.044	CONT	0	15	15	60	10		0	0.66	0.66	2.64	0.44
GRC 10 DATALINK REMOTE CONTROL	N/A	0	CONT	0	15	15	60	10		0	0	0	0	0
GWX 68 RADAR	373	2	CONT	0	5	15	60	5		0	10	30	120	10
GMC 710 A/P MODE CONTROLLER	378	0.183	CONT	0	15	15	60	10		0.00	2.75	2.75	10.98	1.83

Effectivity
4636633, 4636652 and up

CHART 4 (Sheet 5 of 6)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES				
										Start	Taxi	T/O	Cruise	Land
GCU 476 FMS CONTROLLER (KEYPAD)	377	0.14	CONT	0	15	15	60	10		0	2.1	2.1	8.4	1.4
GSA 81 SERVOS (4)	333 / 334	1.6	CONT	0	1.5	1.5	6	1		0	2.4	2.4	9.6	1.6
GEA 71 ENGINE/ AIRFRAME INTERFACE	357	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
GIA 63W #1 INTEGRATED AVIONICS UNIT (TX)	363	4	INT	0	2	1	12	3	INCREASED CURRENT FOR TRANSMIT TIMES ONLY	0	8	4	48	12
GIA 63W #1 INTEGRATED AVIONICS UNIT (RX)	364	1.3	CONT	1	15	15	60	10		1.3	19.5	19.5	78	13
GIA 63W #2 INTEGRATED AVIONICS UNIT (TX)	383	1.3	INT	0	15	15	60	10		0	19.5	19.5	78	13
GIA 63W #2 INTEGRATED AVIONICS UNIT (RX)	382	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
GDC 74A #1 AIR DATA COMPUTER	361	0.23	CONT	1	15	15	60	10		0.23	3.45	3.45	13.8	2.3
GDC 74A #2 AIR DATA COMPUTER	385	0.23	CONT	0	15	15	60	10		0	3.45	3.45	13.8	2.3
GTP 59 #1 OAT PROBE	N/A	0	CONT	1	15	15	60	10	INCLUDED WITH POWER TO ADC	0	0	0	0	0
GTP 59 #2 OAT PROBE	N/A	0	CONT	0	15	15	60	10	INCLUDED WITH POWER TO ADC	0	0	0	0	0
GRS 77 #1 AHRS	362	0.344	CONT	1	15	15	60	10		0.344	5.16	5.16	20.64	3.44
GRS 77 #2 AHRS	384	0.344	CONT	0	15	15	60	10		0	5.16	5.16	20.64	3.44
GMU 44 #1 MAGNETOMETER	N/A	0.3	CONT	1	15	15	60	10		0.3	4.5	4.5	18	3
GMU 44 #2 MAGNETOMETER	N/A	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
GMA 350 Audio Panel	360	0.4	CONT	1	15	15	60	10		0.4	6	6	24	4

Effectivity
4636633, 4636652 and up

CHART 4 (Sheet 6 of 6)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000)

Item	CB POS.	CKT LOAD	INT or CONT	Start Time (1)	Taxi Time (15)	T/O Time (15)	Cruise Time (60)	Land Time (10)	Remarks	AMP-MINUTES Start	Taxi	T/O	Cruise	Land
GTX 33 ES #1 Transponder	374	1	CONT	0	15	15	60	10	Transponder 1 or 2 in operation	0	15	15	60	10
GTX 33 ES #2 Transponder	395	0.85	STBY	0	15	15	60	10	Transponder 1 or 2 in operation	0	12.75	12.75	51	8.5
GAE 43 ALTITUDE DATA SYSTEM	364 / 382	0.1	CONT	0	15	15	60	10		0	1.5	1.5	6	1
Evolution Backup Display (EBD) Indicator	335	2.4	CONT	0	15	15	60	10		0	36	36	144	24
Becker ADF 3500	394	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
KN-63 DME	393	1.5	CONT	0	15	15	60	10		0	22.5	22.5	90	15
WX-500 STORMSCOPE	391	0.38	CONT	0	15	15	60	10		0	5.7	5.7	22.8	3.8
GTS 825 Traffic System	392	1.7	CONT	0	15	15	60	10		0	25.5	25.5	102	17
GSR 56 Iridium	387	0.65	CONT	0	15	15	60	10		0	9.75	9.75	39	6.5
Pulse Oximeter & CO Detector	350	0.3 max 0.05 nom	CONT	0	15	15	60	10		0	4.5	4.5	18	3
Mid-Continent USB charging ports	356	0.85 (3 ea.)	CONT	0	15	15	60	10		0	12.75	12.75	51	8.5
* A/C not used during Night IFR														
									TOTAL AMP MINUTES	77.16	1875.29	1588.76	6101.40	972.90
									UTILIZATION LOAD (AMPS)	77.16	125.02	105.92	101.69	97.29

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Effectivity
4636716, 4636720–4636775

CHART 5 (Sheet 1 of 3)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000 NXi)

ITEM	NO. UNITS	AMPS/ UNIT	OP. CYCLE	Elapsed Time START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Battery Master Relay		0.6	Cont	1	15	15	60	10		0.6	9	9	36	6
Starter		110	Int	0.5	0	0	0	0	Three 10 Sec Engine Crank Cycles	55	0	0	0	0
Starter Contactor		1.47	Int	0.5	0	0	0	0	Same As Engine Starter	0.735	0	0	0	0
Starter Vibrator		2	Int	0.5	0	0	0	0	Same As Engine Starter	1	0	0	0	0
Alt #1 Field & Regulator		1.57	Cont	0	15	15	60	10	1.5 Field And .07 Regulator Current	0	23.55	23.55	94.2	15.7
Alt #2 Field & Regulator		1.57	Cont	0	15	15	60	10	1.5 Field And .07 Regulator Current	0	23.55	23.55	94.2	15.7
Avionics #1 Contactor		0.7	Int	1	0	0	0	0	Avionics On = Deenergized	0.7	0	0	0	0
Avionics #2 Contactor		0.7	Int	1	0	0	0	0	Avionics On = Deenergized	0.7	0	0	0	0
Engine Hour Meter		0.05	Cont	1	15	15	60	10		0.05	0.75	0.75	3	0.5
Left / Right Fuel Boost Pump		4	Cont	1	15	15	60	10	One Pump In Continuous Operation	4	60	60	240	40
Emergency Fuel Pump		4.5	Int	1	0	3	0	5	Start, T/O, Landing & Emergency	4.5	0	13.5	0	22.5
Hydraulic Pump		26	Int	0	0	0.25	0	0.25	During Gear Ext/Ret Only (15 Seconds)	0	0	6.5	0	6.5
Hydraulic Pump Control Relay		0.35	Int	0	0	0.25	0	0.25	During Gear Ext/Ret Only (15 Seconds)	0	0	0.0875	0	0.0875
Stall Warning (Computer)		0.3	Cont	1	15	15	60	10	Safe Flight Computer	0.3	4.5	4.5	18	3
Stall Warning Heat		16.1	Cont	0	15	15	60	10		0	241.5	241.5	966	161
Speed Brakes		0.44	Cont	1	15	15	60	10	2.15 Amps Intermittent At Deploy	0.44	6.6	6.6	26.4	4.4
Electric Flaps		6	Int	0	0.1	0	0	0.25	T/O @ 10 Deg. / Ldg At 36 Deg	0	0.6	0	0	1.5
Flap Warning		0.15	Int	0	0.1	0	0	0.25	Only During Transit	0	0.015	0	0	0.0375
Left Pitot Heat		6	Cont	0	15	15	60	10	50% On Gnd / 100% In Flight	0	45	90	360	60
Right Pitot Heat		6	Cont	0	15	15	60	10	50% On Gnd / 100% In Flight	0	45	90	360	60
Anti-Collision Strobes	2	0.5	Cont	0	0	15	60	10		0	0	15	60	10
Position (Nav) Lights - LED	2	0.17	Cont	0	15	15	60	10		0	5.1	5.1	20.4	3.4
Taxi LED Light	2	0.45	Cont	0	15	0	0	0		0	13.5	0	0	0
Landing LED Light		0.6	Cont	0	0	5	0	5		0	0	3	0	3
Landing Lights Relay		0.7	Cont	0	0	5	0	5	Same As Landing Light	0	0	3.5	0	3.5
Avionics Lights (Dimmer)		1	Cont	0	15	15	60	10	@ 50% Intensity	0	7.5	15	60	10
Instrument Panel Lights / E.L. Placards		1.7	Cont	0	15	15	60	10	Includes Switch Lights (@50% Intensity)	0	25.5	25.5	102	17
Reading Lights (Cabin)	8	0.137	Int	0	1	0	0	0	Normally Off During Flight (5 @ 2 Amps Ea)	0	1.1	0	0	0
Cockpit Flood Light (Map)	2	0.137	Int	0	5	0	0	0	Pre-Take Off Checklist	0	1.4	0	0	0
Vent Defog Blower		7	Cont	0	15	7	0	7		0	105	49	0	49
Cabin Fresh Air Blower	2	6	Cont	0	7.5	7.5	0	0	Not Required In Flight	0	90	90	0	0
Cpcs Ctrl		0.47	Cont	1	15	15	60	10		0.47	7.05	7.05	28.2	4.7
Cabin Pressure Dump Solenoid		0.29	Cont	1	15	0	0	0		0.29	4.35	0	0	0
Vacuum		0.025	Cont	0	15	15	60	10	Vacuum Transducer Only (<25 Ma Load)	0.00	0.38	0.38	1.50	0.25
LH/RH Ametek Capacitance Fuel Quantity		0.07	Cont	0	15	15	60	10		0	1.05	1.05	4.2	0.7
Air Conditioning Control *		2	Int	0	0	0	0	0	Clutch & Relay; Taxi & Climbout	0	0	0	0	0
Air Conditioning Power *		16.5	Int	0	0	0	0	0	Evaporater Blower; Taxi & Climbout	0	0	0	0	0
Propeller De-Ice		18	Cont	0	15	15	60	10	50% Duty Cycle	0	135	135	540	90
Surface De-Ice		1.85	Cont	0	0	15	60	10	50% Duty Cycle	0	0	13.875	55.5	9.25
Windshield Heat Relays (Control)		1.5	Cont	0	15	15	60	10	50% Duty Cycle	0	11.25	11.25	45	7.5
Windshield Heat Low		13.59	Cont	0	15	0	0	0	50% Duty Cycle	0	101.925	0	0	0
Windshield Heat High		20.74	Cont	0	0	15	60	10	50% Duty Cycle	0	0	155.55	622.2	103.7

Effectivity
4636716, 4636720–4636775

CHART 5 (Sheet 2 of 3)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000 NXi)

ITEM	NO. UNITS	AMPS/ UNIT	OP. CYCLE	Elapsed Time START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Wing Ice Light		0.9	Int	0	0.5	0	1	1	Occasional Intermittent Use In Cruise	0	0.45	0	0.9	0.9
Cabin Heater		33	Int	0	15	0	0	0	35 Amp Fuse Off Battery Feed; Used Only	0	495	0	0	0
Cabin Heater Relay		0.35	Int	0	15	0	0	0	During Taxi, Otherwise Off.	0	5.25	0	0	0
Inverter - 110 Volt AC Power		12	Cont	0	0	7.5	60	0	50% Loaded Maximum In Cruise	0	0	90	720	0
Lightning Protection		N/A	Int	0	0	0	0	0	N/A - Only During Lightning Event	0	0	0	0	0
GDU 1050 PFD #1 Heater		1.2	Cont	0	7.5	7.5	30	5	Heater	0	9	9	36	6
GDU 1050 PFD #1		0.7	Cont	1	15	15	60	10		0.7	10.5	10.5	42	7
GDU 1050 PFD #2 Heater		1.2	Cont	0	7.5	7.5	30	5	Heater	0	9	9	36	6
GDU 1050 PFD #2		0.7	Cont	0	15	15	60	10		0	10.5	10.5	42	7
GDU 1250A MFD		1.2	Cont	0	7.5	7.5	30	5	W/ Heaters On	0	9	9	36	6
GDU 1250A MFD		0.7	Cont	0	15	15	60	10		0	10.5	10.5	42	7
Lone Star Av Cyclone 21 Cooling Fan	6	0.19	Cont	0.16	15	15	60	10		0.1824	17.1	17.1	68.4	11.4
GDL 69A Wx Datalink		0.35	Cont	0	15	15	60	10		0	5.25	5.25	21	3.5
GWX 68 Weather Radar		2	Cont	0	5	15	60	5		0	10	30	120	10
GMC 710 A/P Mode Controller		0.16	Cont	0	15	15	60	10		0.00	2.40	2.40	9.60	1.60
GCU 47X FMS Controller		0.14	Cont	1	15	15	60	10	GCU 476 Model	0.14	2.10	2.10	8.40	1.40
GSA 81 Servos (4x)		6.4	Cont	0	3	3	12	2	20% Duty Cycle	0	19.2	19.2	76.8	12.8
GEA 71B Engine Airframe Interface		0.52	Cont	1	15	15	60	10	Max Load	0.52	7.8	7.8	31.2	5.2
GIA 64 #1 Integrated Avionics Unit (Com)		4.3	Int	0	3	3	12	3	Active	0	12.9	12.9	51.6	12.9
GIA 64 #1 Integrated Avionics Unit (Com)		0.3	Cont	1	15	15	60	10	Inactive	0.3	4.5	4.5	18	3
GIA 64 #1 Integrated Avionics Unit (Other)		0.8	Cont	1	15	15	60	10		0.8	12	12	48	8
GIA 64 #2 Integrated Avionics Unit (Com)		4.3	Int	0	0	0	0	0	Active	0	0	0	0	0
GIA 64 #2 Integrated Avionics Unit (Com)		0.3	Cont	0	15	15	60	10	Inactive	0	4.5	4.5	18	3
GIA 64 #2 Integrated Avionics Unit (Other)		0.8	Cont	0	15	15	60	10		0	12	12	48	8
GDC 72 #1 Air Data Computer		0.21	Cont	1	15	15	60	10	Includes GTP 59 Load	0.21	3.15	3.15	12.6	2.1
GDC 72 #2 Air Data Computer		0.21	Cont	0	15	15	60	10	Includes GTP 59 Load	0	3.15	3.15	12.6	2.1
GRS 79 #1 AHRS		0.24	Cont	1	15	15	60	10	Includes GMU 44 Load	0.24	3.6	3.6	14.4	2.4
GRS 79 #2 AHRS		0.24	Cont	1	15	15	60	10	Includes GMU 44 Load	0.24	3.6	3.6	14.4	2.4
GMA 350c Audio Selector Panel		0.4	Cont	1	15	15	60	10		0.4	6	6	24	4
GTX 33D ES (Replaces 335)		1.6	Int	0	7.5	7.5	30	5	Max Full Load	0	12	12	48	8
GTX 33D ES (Replaces 335)		0.85	Int	0	7.5	7.5	30	5	Max Quiescent	0	6.375	6.375	25.5	4.25
GTX 335 Transponder (Standard)		0.43	Int	0	0	0	0	0		0	0	0	0	0
GTX 335 Transponder (Standard)		0.29	Cont	0	0	0	0	0		0	0	0	0	0
GTX 345 Transponder (Optional)		0.65	Int	0	7.5	7.5	30	5	Max Full Load	0	4.875	4.875	19.5	3.25
GTX 345 Transponder (Optional)		0.36	Cont	0	7.5	7.5	30	5	Max Quiescent	0	2.7	2.7	10.8	1.8
GAE 43 Altitude Data System		0.1	Cont	0	15	15	60	10	SAE5-35 Sandia	0	1.5	1.5	6	1
Standby Aspen EFD 1000 Indicator		2.4	Cont	0	15	15	60	10	Includes RSM	0	36	36	144	24
Becker ADF 3500		0.3	Cont	0	15	15	60	10	AD 3502	0	4.5	4.5	18	3
KN-63 DME		0.62	Cont	0	15	15	60	10		0	9.3	9.3	37.2	6.2
WX-500 Stormscope		0.38	Cont	0	15	15	60	10	Max	0	5.7	5.7	22.8	3.8
GTS 825 Traffic System		1.7	Cont	0	15	15	60	10		0	25.5	25.5	102	17
GSR 56 Iridium		0.65	Cont	0	15	15	60	10		0	9.75	9.75	39	6.5

Effectivity
4636716, 4636720–4636775

CHART 5 (Sheet 3 of 3)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000 NXi)

ITEM	NO. UNITS	AMPS/ UNIT	OP. CYCLE	Elapsed Time START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Pulse Oximeter & CO Detector		0.3	Cont	0	15	15	60	10		0	4.5	4.5	18	3
Mid-Continent USB Charging Ports		0.85	Cont	0	15	15	60	10		0	12.75	12.75	51	8.5
* A/C Not Used During Night IFR Icing														
									Total (Amp * Minutes)	72.52	1794.07	1532.99	5830.50	932.93
									Utilization Load (Amps)	72.52	119.60	102.20	97.18	93.29
									Utilization Loads	78.74	146.31	118.73	102.46	102.33
									Percent Change	-7.90	-18.25	-13.92	-5.16	-8.83
									Average Percent Change = -10.81					

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Effectivity
4636776 and up

CHART 6 (Sheet 1 of 3)
ELECTRICAL LOAD ANALYSIS (PA-46-350P M350 G1000 NXi)

				Elapsed Time						AMPS * MINUTES				
ITEM	NO. UNITS	AMPS/ UNIT	OP. CYCLE	START	TAXI	T/O	CRUISE	LAND	REMARKS	START	TAXI	T/O	CRUISE	LAND
Battery Master Relay		0.6	Cont	1	15	15	60	10		0.6	9	9	36	6
Starter		110	Int	0.5	0	0	0	0	Three 10 Sec Engine Crank Cycles	55	0	0	0	0
Starter Contactor		0.62	Int	0.5	0	0	0	0	Same As Engine Starter	0.31	0	0	0	0
Starter Vibrator		2	Int	0.5	0	0	0	0	Same As Engine Starter	1	0	0	0	0
Alt #1 Field & Regulator		1.57	Cont	0	15	15	60	10	1.5 Field And .07 Regulator Current	0	23.55	23.55	94.2	15.7
Alt #2 Field & Regulator		1.57	Cont	0	15	15	60	10	1.5 Field And .07 Regulator Current	0	23.55	23.55	94.2	15.7
Avionics #1 Contactor		0.37	Int	1	0	0	0	0	Avionics On = Deenergized	0.37	0	0	0	0
Avionics #2 Contactor		0.37	Int	1	0	0	0	0	Avionics On = Deenergized	0.37	0	0	0	0
Engine Hour Meter		0.05	Cont	1	15	15	60	10		0.05	0.75	0.75	3	0.5
Left / Right Fuel Boost Pump		4	Cont	1	15	15	60	10	One Pump In Continuous Operation	4	60	60	240	40
Emergency Fuel Pump		4.5	Int	1	0	3	0	5	Start, T/O, Landing & Emergency	4.5	0	13.5	0	22.5
Hydraulic Pump		26	Int	0	0	0.25	0	0.25	During Gear Ext/Ret Only (15 Seconds)	0	0	6.5	0	6.5
Hydraulic Pump Control Relay		0.35	Int	0	0	0.25	0	0.25	During Gear Ext/Ret Only (15 Seconds)	0	0	0.0875	0	0.0875
Stall Warning (Computer)		0.3	Cont	1	15	15	60	10	Safe Flight Computer	0.3	4.5	4.5	18	3
Stall Warning Heat		9.6	Cont	0	15	15	60	10		0	144	144	576	96
Speed Brakes		0.44	Cont	1	15	15	60	10	2.15 Amps Intermittent At Deploy	0.44	6.6	6.6	26.4	4.4
Electric Flaps		6	Int	0	0.1	0	0	0.25	T/O @ 10 Deg. / Ldg At 36 Deg	0	0.6	0	0	1.5
Flap Warning		0.15	Int	0	0.1	0	0	0.25	Only During Transit	0	0.015	0	0	0.0375
Left Pitot Heat		12.14		0	15	15	60	10	50% On Gnd / 100% In Flight	0	91.05	182.1	728.4	121.4
Right Pitot Heat		12.14	Cont	0	15	15	60	10	50% On Gnd / 100% In Flight	0	91.05	182.1	728.4	121.4
Anti-Collision Strobes	2	0.5	Cont	0	0	15	60	10		0	0	15	60	10
Position (Nav) Lights - LED	2	0.17	Cont	0	15	15	60	10		0	5.1	5.1	20.4	3.4
Taxi LED Light	2	0.45	Cont	0	15	0	0	0		0	13.5	0	0	0
Landing LED Light		0.6	Cont	0	0	5	0	5		0	0	3	0	3
Landing Lights Relay		0.7	Cont	0	0	5	0	5	Same As Landing Light	0	0	3.5	0	3.5
Avionics Lights (Dimmer)		1	Cont	0	15	15	60	10	@ 50% Intensity	0	7.5	15	60	10
Instrument Panel Lights / E.L. Placards		1.7	Cont	0	15	15	60	10	Includes Switch Lights (@50% Intensity)	0	25.5	25.5	102	17
Reading Lights (Cabin)	8	0.137	Int	0	1	0	0	0	Normally Off During Flight (5 @ 2 Amps Ea)	0	1.1	0	0	0
Cockpit Flood Light (Map)	2	0.04	Int	0	5	0	0	0	Pre-Take Off Checklist	0	0.4	0	0	0
Vent Defog Blower		7	Cont	0	15	7	0	7		0	105	49	0	49
Cabin Fresh Air Blower	2	6	Cont	0	7.5	7.5	0	0	Not Required In Flight	0	90	90	0	0
Cpcs Ctrl		0.47	Cont	1	15	15	60	10		0.47	7.05	7.05	28.2	4.7
Cabin Pressure Dump Solenoid		0.29	Cont	1	15	0	0	0		0.29	4.35	0	0	0
Vacuum		0.025	Cont	0	15	15	60	10	Vacuum Transducer Only (<25 Ma Load)	0.00	0.38	0.38	1.50	0.25
LH/RH Ametek Capacitance Fuel Quantity		0.07	Cont	0	15	15	60	10		0	1.05	1.05	4.2	0.7
Air Conditioning Control *		2	Int	0	0	0	0	0	Clutch & Relay; Taxi & Climbout	0	0	0	0	0
Air Conditioning Power *		16.5	Int	0	0	0	0	0	Evaporater Blower; Taxi & Climbout	0	0	0	0	0
Propeller De-Ice		18	Cont	0	15	15	60	10	50% Duty Cycle	0	135	135	540	90
Surface De-Ice		1.85	Cont	0	0	15	60	10	50% Duty Cycle	0	0	13.875	55.5	9.25
Windshield Heat Relays (Control)		0.7	Cont	0	15	15	60	10	50% Duty Cycle	0	5.25	5.25	21	3.5
Windshield Heat Low		17.7	Cont	0	15	0	0	0	50% Duty Cycle	0	132.75	0	0	0
Windshield Heat High		24.3	Cont	0	0	15	60	10	50% Duty Cycle	0	0	182.25	729	121.5

Effectivity
4636776 and up

CHART 6 (Sheet 2 of 3)
ELECTRICAL LOAD ANALYSIS - GARMIN G1000 NXi EQUIPPED WITH GFC 700 AUTOPILOT

ITEM	NO. UNITS	AMPS/ UNIT	OP. CYCLE	Elapsed Time START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Wing Ice Light		0.9	Int	0	0.5	0	1	1	Occasional Intermittent Use In Cruise	0	0.45	0	0.9	0.9
Cabin Heater		33	Int	0	15	0	0	0	35 Amp Fuse Off Battery Feed; Used Only	0	495	0	0	0
Cabin Heater Relay		0.37	Int	0	15	0	0	0	During Taxi, Otherwise Off.	0	5.55	0	0	0
True Blue Inverter - 110 Volt Ac Power		10	Cont	0	0	15	60	10	50% Loaded Maximum In Cruise	0	0	150	600	100
Lightning Protection		N/A	Int	0	0	0	0	0	N/A - Only During Lightning Event	0	0	0	0	0
GDU 1050 PFD #1 Heater		1.2	Cont	0	7.5	7.5	30	5	Heater	0	9	9	36	6
GDU 1050 PFD #1		0.7	Cont	1	15	15	60	10		0.7	10.5	10.5	42	7
GDU 1050 PFD #2 Heater		1.2	Cont	0	7.5	7.5	30	5	Heater	0	9	9	36	6
GDU 1050 PFD #2		0.7	Cont	0	15	15	60	10		0	10.5	10.5	42	7
GDU 1250A MFD		1.2	Cont	0	7.5	7.5	30	5	W/ Heaters On	0	9	9	36	6
GDU 1250A MFD		0.7	Cont	0	15	15	60	10		0	10.5	10.5	42	7
Lone Star Av Cyclone 21 Cooling Fan	6	0.19	Cont	0.16	15	15	60	10		0.1824	17.1	17.1	68.4	11.4
GDL 69A Wx Datalink		0.35	Cont	0	15	15	60	10		0	5.25	5.25	21	3.5
GWX 75 Weather Radar		2.5	Cont	0	5	15	60	5		0	12.5	37.5	150	12.5
GMC 710 A/P Mode Controller		0.1	Cont	0	15	15	60	10		0.00	1.50	1.50	6.00	1.00
GCU 47X FMS Controller		0.14	Cont	1	15	15	60	10	GCU 476 Model	0.14	2.10	2.10	8.40	1.40
GSA 81 Servos (4x)		6.4	Cont	0	3	3	12	2	20% Duty Cycle	0	19.2	19.2	76.8	12.8
GEA 71B Engine Airframe Interface		0.52	Cont	1	15	15	60	10	Max Load	0.52	7.8	7.8	31.2	5.2
GIA 64 #1 Integrated Avionics Unit (Com)		4.3	Int	0	3	3	12	3	Active	0	12.9	12.9	51.6	12.9
GIA 64 #1 Integrated Avionics Unit (Com)		0.3	Cont	1	15	15	60	10	Inactive	0.3	4.5	4.5	18	3
GIA 64 #1 Integrated Avionics Unit (Other)		0.8	Cont	1	15	15	60	10		0.8	12	12	48	8
GIA 64 #2 Integrated Avionics Unit (Com)		4.3	Int	0	0	0	0	0	Active	0	0	0	0	0
GIA 64 #2 Integrated Avionics Unit (Com)		0.3	Cont	0	15	15	60	10	Inactive	0	4.5	4.5	18	3
GIA 64 #2 Integrated Avionics Unit (Other)		0.8	Cont	0	15	15	60	10		0	12	12	48	8
GDC 72 #1 Air Data Computer		0.21	Cont	1	15	15	60	10	Includes GTP 59 Load	0.21	3.15	3.15	12.6	2.1
GDC 72 #2 Air Data Computer		0.21	Cont	0	15	15	60	10	Includes GTP 59 Load	0	3.15	3.15	12.6	2.1
GRS 79 #1 AHRS		0.24	Cont	1	15	15	60	10	Includes GMU 44 Load	0.24	3.6	3.6	14.4	2.4
GRS 79 #2 AHRS		0.24	Cont	1	15	15	60	10	Includes GMU 44 Load	0.24	3.6	3.6	14.4	2.4
GMA 350C Audio Selector Panel		0.4	Cont	1	15	15	60	10		0.4	6	6	24	4
GTX 334DR Transponder (Optional)		0.65	Int	0	0	0	0	0		0	0	0	0	0
GTX 345DR Transponder (Optional)		0.36	Cont	0	0	0	0	0		0	0	0	0	0
GTX 335 Transponder (Optional)		0.43	Int	0	7.5	7.5	30	5		0	3.225	3.225	12.9	2.15
GTX 335 Transponder (Optional)		0.29	Cont	0	7.5	7.5	30	5		0	2.175	2.175	8.7	1.45
GTX 345 Transponder (Standard)		0.65	Int	0	7.5	7.5	30	5	Max Full Load	0	4.875	4.875	19.5	3.25
GTX 345 Transponder (Standard)		0.36	Cont	0	7.5	7.5	30	5	Max Quiescent	0	2.7	2.7	10.8	1.8
GAE 43 Altitude Data System		0.1	Cont	0	15	15	60	10	SAE5-35 Sandia	0	1.5	1.5	6	1
GI 275 Standby Instrument		1	Cont	0	15	15	60	10	Includes Gmu 11	0	15	15	60	10
Becker ADF 3500		1.1	Cont	0	15	15	60	10	AD 3502	0	16.5	16.5	66	11
KN-63 DME		0.62	Cont	0	15	15	60	10		0	9.3	9.3	37.2	6.2
WX-500 Stormscope		0.38	Cont	0	15	15	60	10	Max	0	5.7	5.7	22.8	3.8
GTS 825 Traffic System		1.7	Cont	0	15	15	60	10		0	25.5	25.5	102	17
GSR 56 Iridium		1.86	Cont	0	15	15	60	10		0	27.9	27.9	111.6	18.6

Effectivity
4636776 and up

CHART 6 (Sheet 3 of 3)
ELECTRICAL LOAD ANALYSIS - GARMIN G1000 NXi EQUIPPED WITH GFC 700 AUTOPILOT

ITEM	NO. UNITS	AMPS/ UNIT	OP. CYCLE	Elapsed Time START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Pulse Oximeter & CO Detector		0.3	Cont	0	15	15	60	10		0	4.5	4.5	18	3
Mid-Continent USB Charging Ports		0.85	Cont	0	15	15	60	10		0	12.75	12.75	51	8.5
* A/C Not Used During Night Ifr Icing														
									Total (Amp * Minutes)	71.43	1810.57	1703.16	6151.20	1103.88
									Utilization Load (Amps)	71.43	120.70	113.54	102.52	110.39
									Utilization Loads	78.74	146.31	118.73	102.46	102.33
									Percent Change	-9.28	-17.50	-4.37	0.06	7.87
									Average Percent Change = -4.64					

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CHAPTER

25

EQUIPMENT / FURNISHINGS

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CHAPTER 25

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
25-List of Effective Pages	1	Mar 25/21	25-50-00	1	Sep 15/09
	2	Sep 15/09		2	Sep 15/09
25-Table of Contents	1	Nov 30/17	25-60-00	1	Nov 30/17
	2	Mar 25/21		2	Nov 30/17
	3	Mar 25/21		3	Nov 30/17
	4	Jan 15/13		4	Nov 30/17
				5	Nov 30/17
25-10-00	1	Sep 15/09		6	Nov 30/17
	2	Jul 1/10		7	Nov 30/17
	3	Nov 30/17		8	Nov 30/17
	4	Dec 15/14		9	Nov 30/17
	5	Sep 15/09		10	Nov 30/17
	6	Nov 30/17		11	Nov 30/17
	7	Nov 30/17		12	Nov 30/17
	8	Nov 30/17		13	Nov 30/17
	9	Nov 30/17		14	Nov 30/17
	10	Nov 30 17		15	Nov 30/17
	11	Nov 30 17		16	Nov 30/17
	12	Jan 15/13		17	Nov 30/17
	13	Jan 15/13		18	Nov 30/17
	14	Nov 30/17		19	Mar 25/21
	15	Nov 30/17		20	Nov 30/17
	16	Jan 15/13		21	Mar 25/21
				22	Nov 30/17
25-20-00	1	Nov 30/17		23	Nov 30/17
	2	Jan 15/13		24	Nov 30/17
	3	Dec 15/14		25	Nov 30/17
	4	Dec 15/14		26	Nov 30/17
	5	Nov 30/17		27	Nov 30/17
	6	Jan 15/13		28	Mar 25/21
	7	Jan 15/13		29	Mar 25/21
	8	Jan 15/13		30	Mar 25/21
	9	Jan 15/13			
	10	Nov 30/17			
	11	Nov 30/17			
	12	Nov 30/17			
	13	Nov 30/17			
	14	Jan 15/13			
	15	Nov 30/17			
	16	Nov 30/17			

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CHAPTER 25 - EQUIPMENT / FURNISHINGS

TABLE OF CONTENTS

CHAPTER 25 - EQUIPMENT / FURNISHINGS

TABLE OF CONTENTS (continued)

CHAPTER 25 - EQUIPMENT / FURNISHINGS

TABLE OF CONTENTS (continued)

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FLIGHT COMPARTMENT

1. Pilot's / Copilot's Vertically Adjustable Seat

NOTE: In PA-46-350P S/N's 4636001 thru 4636375, installation of Piper Kit No. 767-522 will extend the service life of the Crew Seat Adjustment Control Cable.

A. Removal

(1) Remove front stops from seat tracks.

(2) Lift lever underneath front lip of seat.

(3) Slide seat off of tracks.

B. Installation

CAUTION: WITH SEAT OCCUPIED AND AT LOWEST VERTICAL POSITION, CHECK SEAT FOR PROPER ENGAGEMENT OF PINS TO SEAT TRACK.

(1) Lift lever under front lip of seat and slide seat onto tracks.

(2) Install front stops on seat tracks.

C. Disassembly

(1) To remove the armrest from the seat back:

(a) Reach up inside seat and carefully remove spring from armrest pivot bar.

(b) Remove bushing from armrest pivot bar.

(c) Slide armrest and pivot bar assembly from seat back.

(2) To remove the seat back assembly from the seat assembly:

(a) Remove nuts, bolts and washers from upper part of seat back support.

(b) Remove seat back.

(3) To remove seat track plunger cable:

(a) Remove cotter pin, washer and pin that secures cable yoke to track plunger actuating lever.

(b) Remove the clamps which secure the upper portion of the cable sheath to the bottom of the seat frame.

(c) Remove the cotter pin, washer and pin that secure the remaining end of the cable to the plunger actuating arm.

(d) Remove the two clamps which secure the cable sheath end to the mount on the lower seat frame cross member.

(4) To remove the gas spring cylinder from the seat:

(a) Remove the two bolts and washers that secure the upper end of the gas spring cylinder to the adapter on the seat frame.

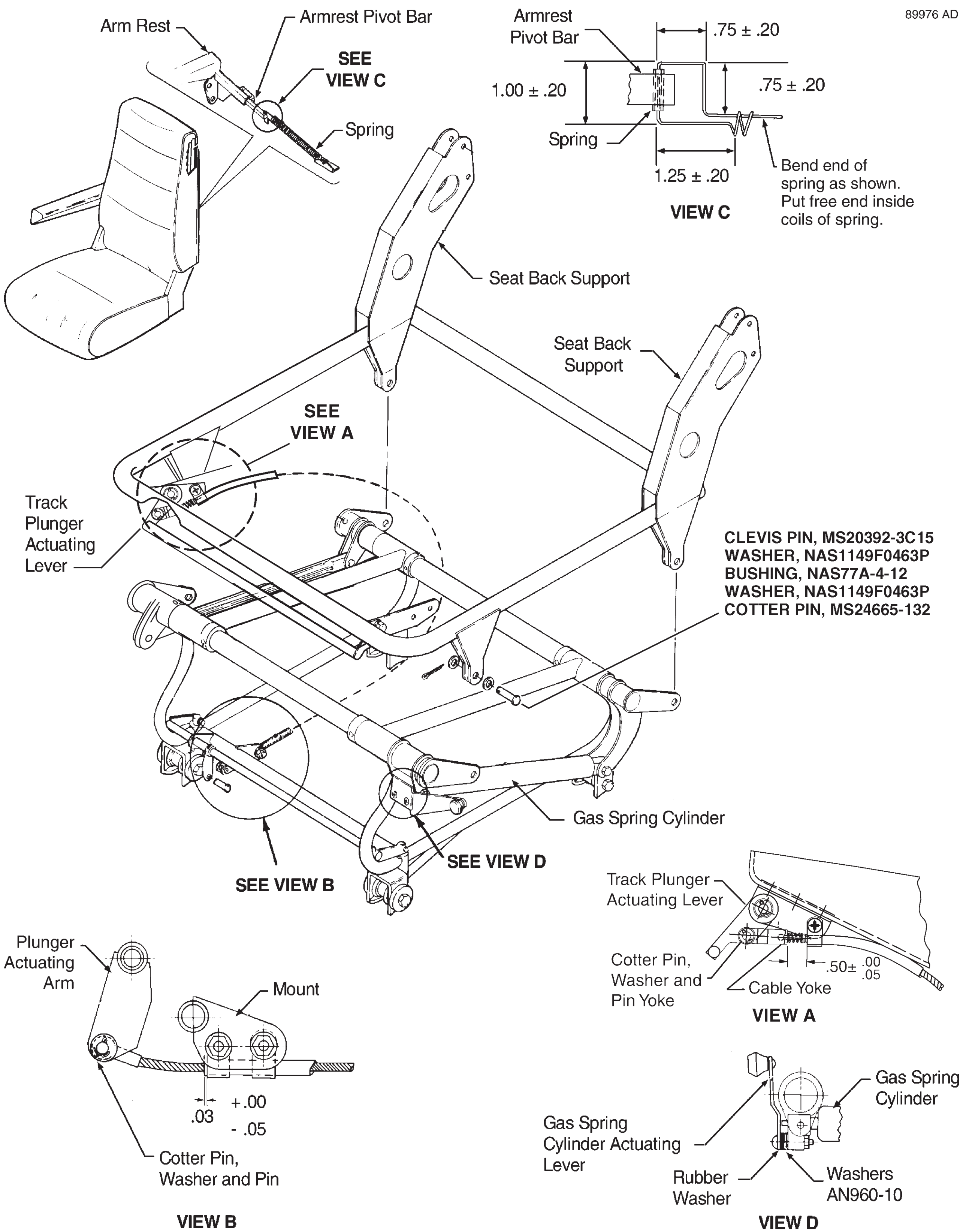
(b) Remove the nut, bolt and washer that secures the lower end of the gas spring cylinder to the rear seat tube assembly.

D. Assembly

(1) To install the gas spring cylinder in the seat:

(a) Attach the lower end of the gas spring cylinder to the rear seat tube assembly with nut, bolt and washer.

(b) Attach the upper end of the gas spring cylinder to the adapter on the seat frame with two bolts and washers.



Pilot's / Copilot's Vertically Adjustable Seats (Typical)
Figure 1

(2) To install the seat track plunger cable:

(a) Attach the cable sheath end to the mount on the lower seat frame cross member with the two clamps. Ensure that the cable end protrudes .03 (+.00/-.05) inch from the end of the forward clamp. (See View B of “Figure 1”)

(b) Attach the lower end of the cable to the plunger actuating arm with the pin, washer and cotter pin.

(c) Attach the upper end of the cable sheath to the bottom of the seat frame with the clamp. Ensure that the cable end protrudes .50 (+.00/-.05) inch from the clamp. (See View A of “Figure 1”)

(d) Attach the cable yoke to the track plunger actuating lever with the pin, washer and cotter pin.

(3) To install the seat back assembly:

(a) Position seat back in place.

(b) Secure seat back with bolts, nuts and washers inserted in upper part of hinge.

(4) To attach the armrest to the seat back:

(a) Slide armrest and pivot bar assembly into seat back.

(b) Place bushing into pivot bar.

(c) Attach spring to armrest pivot bar.

E. Rigging

After installation of the seat track plunger cable, verify that when the handle assembly is in the normal (Hands Off) position the seat plunger should be full down and locked into the seat track. When the handle assembly is operated, the seat plungers must retract sufficiently into the seat assembly to allow fore and aft movement of the seat along the seat tracks.

2. <u>Lumbar Support</u>

Refer to “Figure 2” on page 25105.

A. Description

The pilot and co-pilot seats incorporate a Lumbar support feature. The installation consists of an inflatable bladder attached to the seat back filler and a inflation bulb located under and on the inboard side of each pilot and co-pilot seat.

B. Removal

To remove the Lumbar bladder for repair or replacement:

(1) Remove seat from airplane.

(2) Loosen velcro securing seat back filler cover.

(3) Remove only enough of seat back filer cover to expose lumbar bladder.

<u>NOTE</u>: Inflation tube may be removed before or after bladder is removed from seat back filler. Tube is not glued to nipple attachment; it can be removed by carefully pulling on tube.

(4) Remove inflation tube from bladder.

CAUTION: DO NOT USE A CHEMICAL SOLVENT TO REMOVE BLADDER. SOLVENT MAY DAMAGE SEAT BACK FILLER

CAUTION: TO AVOID OR MINIMIZE DAMAGE TO SEAT BACK FILLER DURING REMOVAL, USE ONE HAND TO RETAIN SEAT BACK FILLER IN PLACE, WHILE GENTLY REMOVING BLADDER WITH OTHER HAND.

(5) Starting at either right or left edge of bladder, carefully and slowly pull bladder and pad assembly from seat back filler.

C. Assembly

To assemble pad and bladder:

CAUTION: CARE MUST BE TAKEN WHEN USING THE APPROVED SOLVENTS AS THEY HAVE A LOW FLASH POINT. SOLVENTS SHOULD NOT BE USED IN CLOSE AREAS WHERE APPLICATOR CANNOT GET SUFFICIENT FRESH AIR FOR BREATHING.

NOTE: Acetone cleans well but evaporates so fast that it is difficult to wipe dry before solvent evaporates.

(1) Surfaces to be joined shall be clean, dry, and oil free. Abrade rubber or plastic surfaces lightly with Scotch Brite™ or fine sand paper and then clean with solvent again.

(a) Surfaces shall be thoroughly cleaned by wiping with a clean cloth wet with solvent followed by wiping dry with a clean cloth before the solvent evaporates. The cleaning operation shall be repeated using a new clean cloth until all soil and grease is removed as evidence by a clean cloth after wiping. Clean cloths and clean solvent shall be used in the cleaning operation.

(b) APPROVED SOLVENTS:

	Acetone	193-035
	Citra-Safe	279-112
	Safe Care	193-051
	Axalta (Formerly Dupont) 2319S/Plastic Prep.	279-081
	Isopropyl Alcohol	179-918

(2) Apply a layer of 3M Scotch Grip 2210 or Delta Laboratories Contact Adhesive B-10161 (P/N 179-853) cement to smooth side of bladder pad.

(3) Apply a layer of 3M Scotch Grip 2210 or Delta Laboratories Contact Adhesive B-10161 (P/N 179-853) cement to back side of bladder (side away from inflation tube nipple).

(4) Allow the adhesive to dry for 15 minutes minimum, 60 minutes maximum.

NOTE: If the adhesive is allowed to dry more than 60 minutes, another coat may be applied to one of the surfaces, and then bonded after 15 minutes minimum. Both surface do not need to be tacky to achieve a bond.

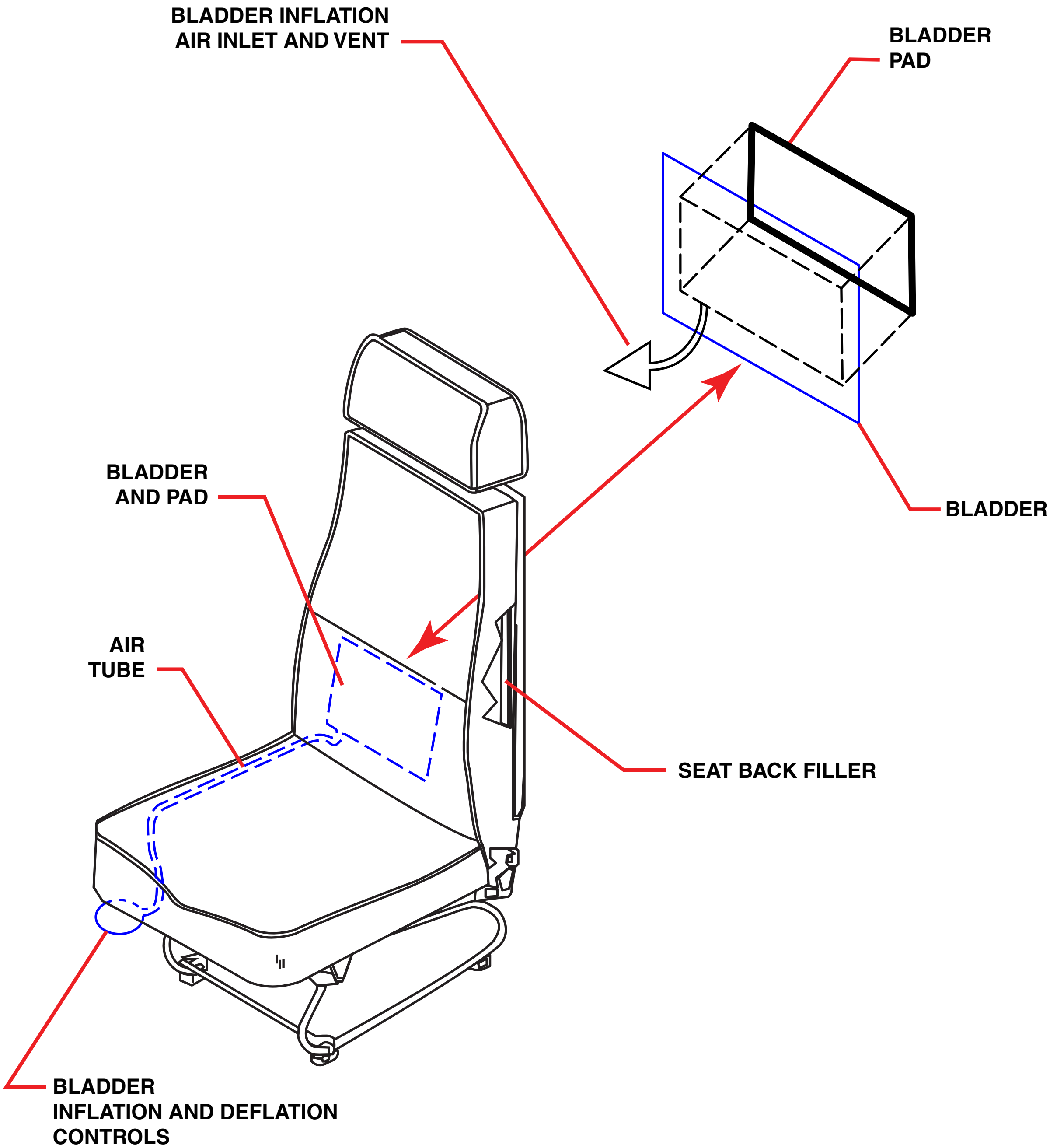
(5) Attach bladder pad to bladder.

(a) Assemble materials when adhesive is dry to touch, but tacky under slight pressure. Position the surfaces to be joined carefully before allowing adhesive to adhesive contact.

NOTE: Repositioning after contact will not be possible without destroying the bond. If this occurs, the adhesive shall be removed and the process repeated.

(b) Press firmly into position over the entire area of the bond.

NOTE: While cement does not set immediately, there is no need to wait before attaching bladder and pad to seat back filler.



Lumbar Seat Bladder Installation
Figure 2

D. Installation

To install Lumbar bladder:

CAUTION: CARE MUST BE TAKEN WHEN USING THE APPROVED SOLVENTS AS THEY HAVE A LOW FLASH POINT. SOLVENTS SHOULD NOT BE USED IN CLOSE AREAS WHERE APPLICATOR CANNOT GET SUFFICIENT FRESH AIR FOR BREATHING.

NOTE: Acetone cleans well but evaporates so fast that it is difficult to wipe dry before solvent evaporates.

(1) Surfaces to be joined shall be clean, dry, and oil-free. Abrade rubber or plastic surfaces lightly with Scotch Brite™ or fine sand paper and then clean with solvent again.

(c) Surfaces shall be thoroughly cleaned by wiping with a clean cloth wet with solvent followed by wiping dry with a clean cloth before the solvent evaporates. The cleaning operation shall be repeated using a new clean cloth until all soil and grease is removed as evidence by a clean cloth after wiping. Clean cloths and clean solvent shall be used in the cleaning operation.

(d) APPROVED SOLVENTS:

	Acetone	193-035
	Citra-Safe	279-112
	Safe Care	193-051
	Axalta (Formerly Dupont) 2319S/Plastic Prep.	279-081
	Isopropyl Alcohol	179-918

(2) Apply a layer of 3M Scotch Grip 2210 or Delta Laboratories contact adhesive B-10161 (P/N 179-853) cement to rough side of bladder pad.

(3) Apply a layer of 3M Scotch Grip 2210 or Delta Laboratories contact adhesive B-10161 (P/N 179-853) cement to seat back filler where bladder is to be located.

(4) Allow the adhesive to dry for 15 minutes minimum, 60 minutes maximum.

NOTE: If the adhesive is allowed to dry more than 60 minutes, another coat may be applied to one of the surfaces, and then bonded after 15 minutes minimum. Both surface do not need to be tacky to achieve a bond.

(5) Assemble materials when adhesive is dry to touch, but tacky under slight pressure. Position the surfaces to be joined carefully before allowing adhesive to adhesive contact.

NOTE: Repositioning after contact will not be possible without destroying the bond. If this occurs, the adhesive shall be removed and the process repeated.

(6) Press firmly into position over the entire area of the bond.

(7) Install seat back filler cover and secure velcro fastenings.

(8) Install seat in airplane.

3. Carpets

The carpets are individually fastened to the floor with hook and loop fasteners and adhesive (Hysol EA9309 NA).

4. Interior Panels - Cockpit

A. General

The aircraft interior includes overhead, wall (side) and aft panels that serve aesthetic and functional purposes. The panels are composed of lightweight fiberglass and covered with soft fabric. Some panel assemblies include electrical terminals or other attachments.

Interior panels are interlocking and, accordingly, must be removed and installed in order. The general rules are:

(1) Remove them from the aft end of the cabin to the front of the cockpit. Remove headliners before side or window panels.

(2) Install them from the front of the cockpit to the aft end of the cabin. Install side or window panels before headliners.

NOTE: Use two people to handle the interior panels while removing and installing them, when practical. While not heavy, the panels can be unwieldy.

NOTE: Each shoulder harness threads through an opening in a side or window panel, requiring that at least the lower outboard attachment of the restraint system must be separated from its attachment point and the belt buckle unlatched to remove an associated side or window panel.

B. Removal

(1) Cockpit Headliner

See “Figure 3” on page 25108

(a) Remove the Baggage Closeout and Cabin Headliner Panels as described in 25-20-00.

(b) Remove the Aft Side and Cabin Window Side Panels as described in 25-20-00.

(c) The cockpit headliner is attached to fuselage supports by two screws and washers at its aft end, remove them first.

(d) On later aircraft, there may be two screws fastening the outside forward edge of the headliner to fuselage supports, remove them as well.

(e) Pull loose the Hook and loop fasteners now holding the headliner up. Support the headliner as it comes free.

(f) Disconnect all electrical connectors for the lights and switch panel.

(g) Remove the cockpit headliner panel and set it, and its hardware, aside in a secure area.

(2) Cockpit Side Window Panels

(a) Remove the Baggage Closeout and Cabin Headliner Panels as described in 25-20-00.

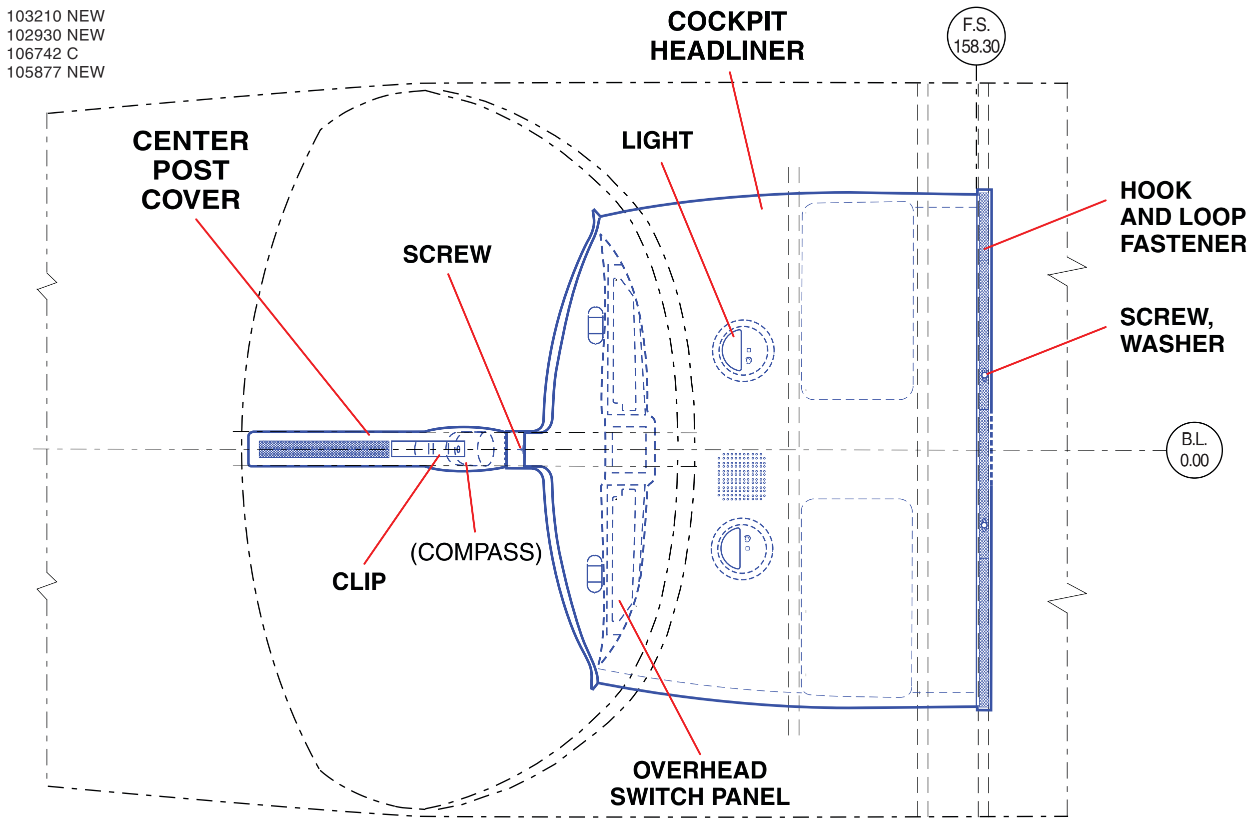
(b) Remove the Cockpit Headliner as described above.

(c) Unbuckle the associated seat belt and detach the lower outboard attaching point of the seat belt / shoulder harness per Restraint System, Removal, below.

(d) Remove the buttons and underlying screws for the seat belt bezel in the panel assembly. Remove the plastic bezel and inner plate, then slip them off of the belt.

(e) Begin to detach the window panel. It is attached to the fuselage by hook and loop fasteners and attached by retainers to the lower cabin panel (see 25-20-00, “Figure 2”, View 6).

(f) Loosen the panel, or remove the panel entirely, and hold it securely. Carefully disconnect electrical and vent attachments at the back of the panel. (See 25-20-00, “Figure 2”, Detail A.)



Cockpit Headliner and Center Post Panels
Figure 3

(g) Hold the panel securely and separate the safety belts from the panel, as follows:

CAUTION: CAREFULLY GUIDE THE METAL BUCKLE AND END FITTINGS THROUGH THE BELT OPENING.

1) Pull the belt assembly through the panel's belt opening, toward the reel (i.e., from the finished side to the back side of the panel).

2) Carefully guide the metal buckle and end fittings through the belt opening.

3) Once the belt is free from the panel, draw the belt around the crew seat securely.

(h) Remove the panel from the aircraft. Place it in a clean, secure area.

C. Installation

(1) Cockpit Side Window Panels

See "Figure 2" on page 25206

NOTE: The crew shoulder harness threads through an opening in this panel. Ensure that the upper attachments and inertia reel plates are installed and the inertia reel properly adjusted per "C. Installation" under Restraint System, below, before attempting to install this panel.

(a) Position the panel in the aircraft.

(b) Hold the panel securely throughout the following procedure.

(c) Thread the lower outboard attaching end of the crew safety belt through the panel, as follows:

CAUTION: CAREFULLY GUIDE THE METAL BUCKLE AND END FITTINGS THROUGH THE BELT OPENING.

1) Push the belt assembly through the panel's belt opening, from the back side of the panel to finished side.

2) Once the end fittings and metal buckle are through the panel, draw the remaining belt through as well.

3) Wrap the belt loosely behind the crew seat headrest for security and to get it out of the way for the remainder of the panel installation.

(d) Carefully connect electrical and vent attachments at the back of the panel. See "Figure 2" on page 25206, Detail A1.

(e) Attach the panel retainers (see 25-20-00, "Figure 2", View 6) into the lower side panel.

(f) Press the panel against the fuselage to join the hook and loop fasteners, confirming that the panel shape aligns with the fuselage shape.

(g) Slip the end fittings and metal buckle through the inner plate and plastic shoulder belt bezel and then slide them up the belt to the panel.

(h) Install the screws and buttons to secure the shoulder belt bezel to the panel.

(i) As required:

1) Install the Cabin Window Side and Aft Side Panels as described in 25-20-00.

2) Install the Cockpit Headliner as described below.

3) Install the Cabin Headliner and Baggage Closeout Panels as described in 25-20-00.

(2) Cockpit Headliner

See "Figure 3".

The cockpit headliner includes the overhead switch panel, a speaker and lighting. It is attached over the interlocking edges of the cockpit side window panels and center post cover panel.

(a) Install cockpit side window panels as described above.

(b) Holding the headliner securely, place it adjacent to its final installed position.

(c) Connect all electrical connectors for the lights and switch panel.

(d) Install the headliner:

1) Attach it to fuselage supports by two screws at its aft end.

2) At its forward point, attach headliner to the center post (with the cover panel) with the provided screw.

3) Hook and loop fasteners attach the left and right side, interlocked over the side window panels.

4) On later aircraft, install two screws fastening the outside forward edge of the headliner to fuselage supports.

(e) As required:

1) See "Cabin Side Window Panels" on page 252010 and "Aft Side Panels" on page 252011 for installation.

2) See "Cabin Headliner" on page 252012 and "Baggage Closeout Panel" on page 252012 for installation.

5. Restraint System

An integrated shoulder harness / lap belt restraint system is installed in PA-46-350P S/N's 4636248 & up and all PA-46R-350Ts. Earlier airplanes employ a traditional lap belt with a separate, detachable, shoulder harness.

The integrated shoulder harness / lap belt restraint system installed in PA-46-350P S/N's 4636248 & up and all PA-46R-350Ts ("Figure 4") is installed for each seat ("Figure 5" on page 251012). They each include a lap belt, shoulder harness, inertia reel, buckle, clasp assembly, end fittings, and attachments. The restraint system upper attachments and inertia reels are located behind interior panels (see "Figure 5" on page 251012).

To remove, install or adjust the restraints, several lightweight, fabric-covered interior panels must be loosened or removed. The panels are interlocking and must be removed and installed in a specific order, see "Interior Panels - Cockpit", General, above.

NOTE: An optional AmSafe Inflatable Seat Restraint System is available to replace the standard pilot and co-pilot seat belt assemblies. When installed, maintain it per AmSafe Instructions for Continued Airworthiness (ICA).

A. Inspection

(1) In S/N's 4636001 thru 4636247 only:

(a) Shoulder Harness

1) Inspect ends and attachment points for condition and security.

2) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of inertial reel. If excessively worn, replace.

3) Check inertia reel mechanism by pulling sharply on strap. Verify reel will lock in place under sudden stress.

(b) Lap Belt

1) Inspect ends and attachment points for condition and security.

2) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of adjustable buckle end. If excessively worn, replace.

3) Inspect shoulder harness keeper nylon bushing. If excessively worn or missing, replacement of that half of the lap belt is required.

(c) Additionally, in S/N's 4636001 thru 4636011 only:

1) Verify compliance with Piper Service Bulletin No. 990. If not yet complied with, then inspect seat belts at each position, as follows:

a) Replace all belts manufactured by Pacific Scientific and the seat belt has a date stamp of 6/94 thru 8/95, if the seat belt buckle end has an old Piper logo (large stylized "P") or no logo, and if the seat belt fitting end I.D. tag has a date stamp of 6/94 thru 8/95.

b) Replacement belts, fitting end and buckle end, should be ordered under the same part number(s) found on the suspect seat belts as no new part numbers were issued. Order the appropriate quantity of: Seat Belt, Fitting End - P/N 564- 867; Buckle End (short) - P/N 564-868; Buckle End (medium) - P/N 564-889; and/or, Buckle End (long) - P/N 564-862.

c) Check the new seat belts for operation and security, after installation.

d) Make an appropriate logbook entry of compliance with Service Bulletin No. 990.

2) Pending replacement of the suspect seat belts, the aircraft may be flown, provided the Pilot-In-Command inspects each seat belt being used for secure positive latching and that the seat belts are free of any anomalies prior to each flight and during each use.

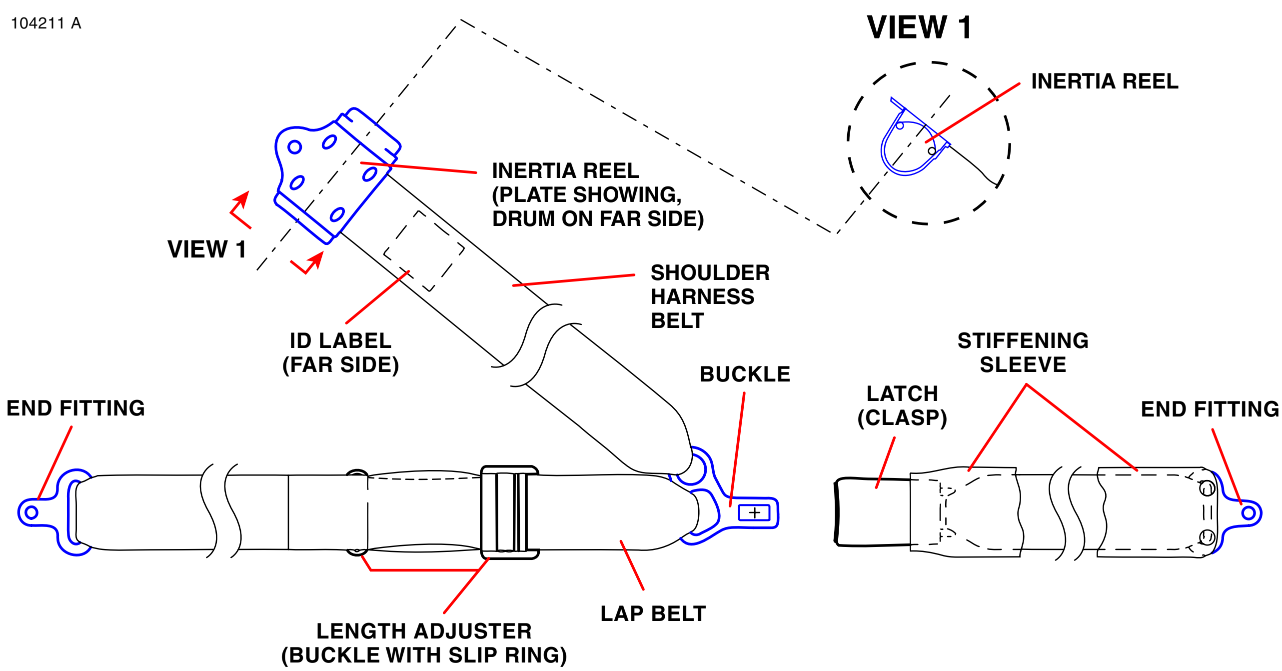
(2) In PA-46-350P S/N's 4636248 & up and all PA-46R-350Ts:

Integrated Shoulder Harness / Lap Belt (Schroth)

(a) Inspect ends and attachment points for condition and security.

(b) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of inertial reel and in and out of the adjusting buckle. If excessively worn, replace.

(c) Check inertia reel mechanism by pulling sharply on strap. Verify reel will lock in place under sudden stress.



Restraint System Belt Assembly (Typical)
Figure 4

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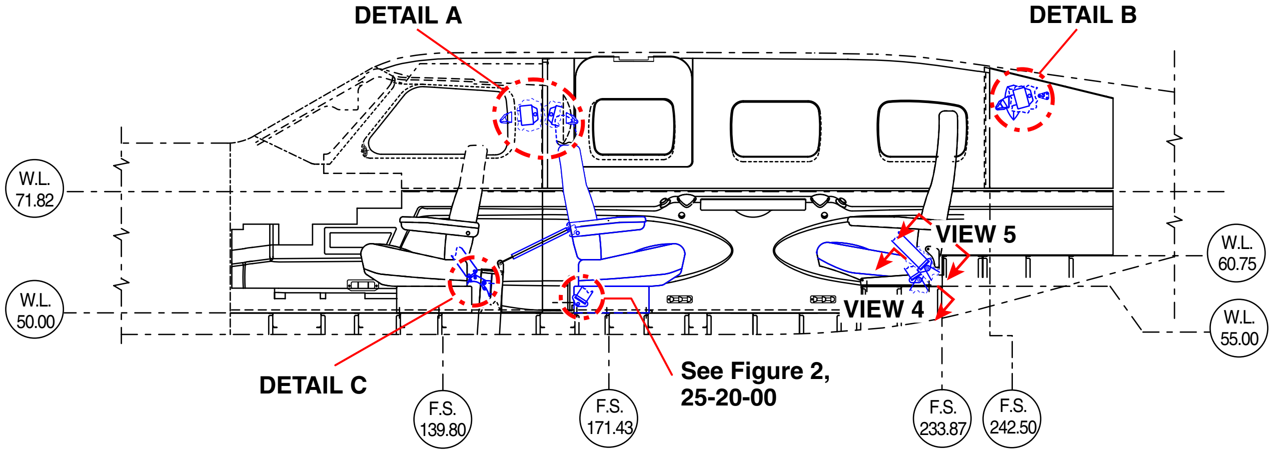
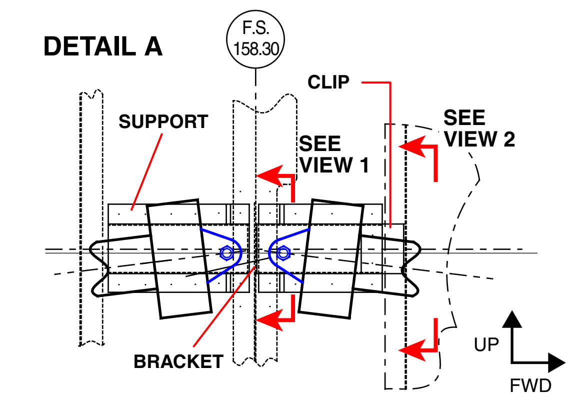
COCKPIT AND CABIN SEAT RESTRAINTS

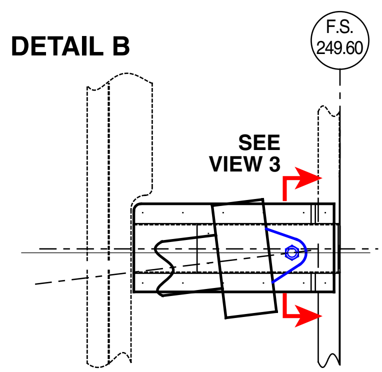
RH SIDE VIEW - LOOKING OUTBOARD



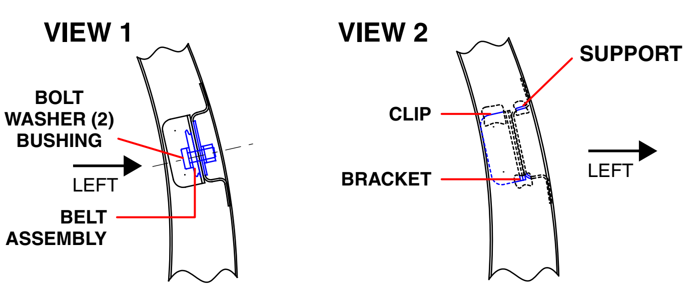
UPPER ATTACHMENTS



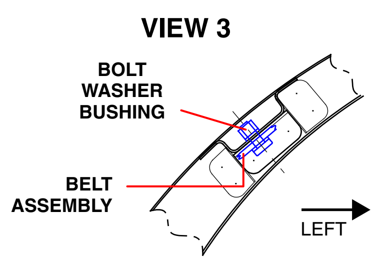
COCKPIT AND AFT-FACING SEAT ATTACHMENTS
LH SIDE - LOOKING OUTBOARD
(RH OPPOSITE)



BENCH ATTACHMENT
RH SIDE - LOOKING OUTBOARD
(LH OPPOSITE)



CREW AND AFT-FACING SEAT
RESTRAINT ATTACHMENTS
LH SIDE - LOOKING AFT
(RH OPPOSITE)

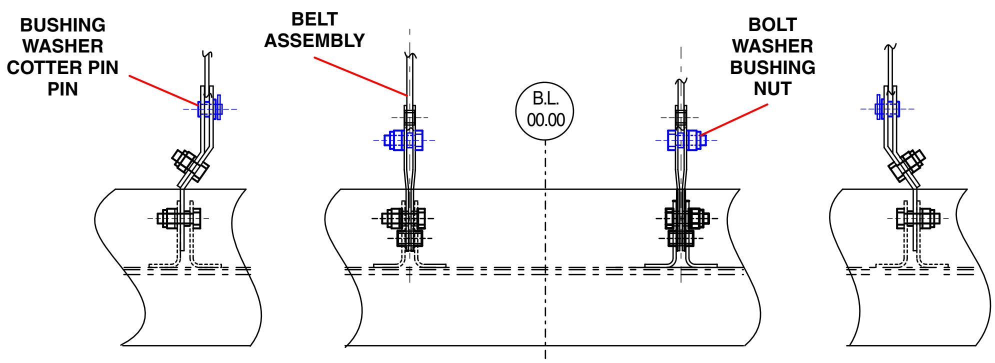


BENCH
RESTRAINT ATTACHMENT
RH SIDE - LOOKING AFT
(LH OPPOSITE)

Restraint System
Figure 5 (Sheet 1 of 2)

LOWER ATTACHMENTS

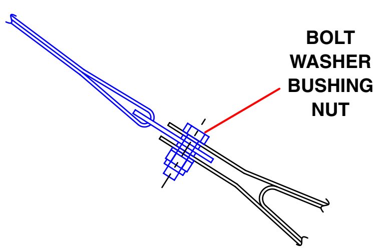
DETAIL C
CREW SEAT RESTRAINTS



INBOARD AND OUTBOARD ASSEMBLIES
LOOKING DOWN

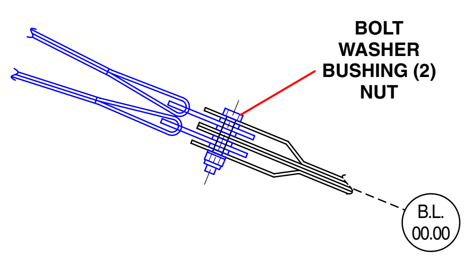
BENCH RESTRAINTS

VIEW 4



OUTBOARD ATTACHMENTS
RH SIDE SHOWN
LOOKING DOWN
(LH OPPOSITE)

VIEW 5



INBOARD ATTACHMENTS
LOOKING DOWN

Restraint System
Figure 5 (Sheet 2 of 2)

B. Removal

The crew restraint system upper attachments and inertia reels are located behind the cockpit side window panels (see “Figure 5” on page 251012). To service or remove the reels or upper attachments, you must remove the cockpit side window panels per Interior Panels - Cockpit, “Removal” on page 25107.

Remove the lower end fittings first.

NOTE: Removal procedures for passenger seat restraints are in Restraint System - Cabin, “Removal” on page 252013.

(1) Lower End Fitting Attachments

See “Figure 5” on page 251013, Detail C.

The lower end fittings of the crew restraint assemblies are attached to links or clevises, which are attached to the floor structure.

Remove the lower end fittings from the link and clevis assemblies at either side of the seat as follows:

(a) Unlatch the seat belt buckle.

(b) Remove the bushing assemblies that attach the outboard end fittings to clevis assemblies.

(c) If required, remove the bolt assemblies that attach the inside (aisle) end fittings that secure the belt latches and stiffening sleeves to the link assemblies.

(2) Belt Assemblies and Cockpit Side Window Panels

See “Figure 2” on page 25206.

Remove the cockpit side window panels per Interior Panels - Cockpit,, “Removal” on page 25107.

(3) Upper Attachments and Inertia Reels

See “Figure 5” on page 251012.

The inertia reel is attached to a metal plate, which is in turn attached to a bracket. The bracket is attached to a fuselage support. (See “Figure 5” on page 251012, Sheet 1.)

To remove the upper attachments and inertia reel plates, remove the bolt assembly from the support bracket.

C. Installation

As with removal, installation of the crew restraint system necessarily involves interior panels. To install the crew integrated restraint system, proceed as follows:

NOTE: Installation procedures for passenger seat restraints are in Restraint System - Cabin, “Installation” on page 252016.

(1) Upper Attachments and Inertia Reel Plates

See “Figure 4” on page 251011 and “Figure 5” on page 251012.

Install the restraint system upper attachments with bolt assemblies (see “Figure 5” on page 251012, Sheet 1) to bracket supports.

NOTE: As necessary, adjust the restraint system inertia reels before installing the interior panels. To adjust the inertia reels, see “Inertial Reel Adjustment” on page 251015.

(2) Belt Assemblies and Cockpit Side Window Panels

See “Figure 2” on page 25206.

Install the cockpit side window panels per Interior Panels - Cockpit, “Installation” on page 25108 .

(3) Lower End Fitting Attachments

See "Figure 5" on page 251013.

Attach the lower end fittings to the link and clevis assemblies at either side of the seat as follows:.

(a) If removed, attach the bolt assemblies to the inside (aisle) end fittings to the link assemblies to secure the belt latches and stiffening sleeves.

(b) Install the bushing assemblies that attach the outboard end fittings to clevis assemblies.

(c) Install the seat belt buckle to the belt latch.

D. Inertial Reel Adjustment

The inertial reel locking feature prevents the shoulder strap from extending and holds occupant in place. For normal movement, strap will extend and retract as required. If required, adjust inertial reel as follows:

(1) Allow harness to wind up on reel as much as possible.

(2) On end of reel, pry off plastic cover over spring. Make sure spring does not come out of plastic cover. Set aside plastic cover.

(3) Unwind the harness completely. Measure and mark the harness 24 inches from the reel center.

(4) Wind harness onto reel until the 24 inch mark is reached. Hold reel and place cap with spring over reel shaft end.

(5) Align slot in shaft with spring tang. Wind spring 6 1/2 turns and snap plastic cover into holes in reel end shaft.

(6) Release harness and allow harness to wind up. Extend harness several times to check reel for smooth operation.

(7) Hold inertia reel with reel completely wound and inertia mechanism end up. Pry off plastic cover over mechanism and set reel aside.

(8) Install nut in plastic cover so that stud in cover is flush with nut surface. Position cover over reel and snap cover into place. Extend harness several times to ensure reel operates smoothly.

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PASSENGER COMPARTMENT

1. Aft Facing Passenger Seats

NOTE: The restraint system lower end attachments are attached to the pedestals of the aft-facing seats.

See “Figure 1” on page 25202.

A. Removal

(1) Disconnect the upper end of the gas spring support from the bracket on the outboard side of the seat back.

(2) If removing or replacing the restraint system, remove the restraint system belt latch and stiffening sleeve assembly (See “Figure 4” on page 252015, Detail D) from the inboard side of the seat pedestal.

(3) Pull off the seat bottom cushion, which is attached to the seat bottom assembly with hook and loop fasteners. Set aside cushion in a clean location.

(4) Remove the eight (8) screws that fasten the seat bottom assembly to the pedestal. Remove the seat bottom assembly and set aside in a clean location.

(5) Remove the six (6) bolt assemblies that secure the seat pedestal to the floor, located along the bottom inner frame. (See “Figure 4” on page 252014, View 3.)

(6) Slide the seat assembly toward the center of the cabin.

(7) Remove the bolt which attaches the outboard lower end fitting of the restraint system to the seat pedestal. (See “Figure 4” on page 252015, Detail E.)

(8) Remove the entire seat assembly from the aircraft.

B. Installation

(1) Place partially assembled seat in the center of the cabin.

(2) Attach the restraint system outboard lower attachment to the seat pedestal.

See “Figure 4” on page 252015, Detail E.

(3) Place the seat pedestal in its floor installation position.

(4) Secure the pedestal to the floor with six (6) bolts and washers.

(5) Secure the seat bottom assembly to the seat pedestal with eight (8) screws.

(6) Insert the seat bottom cushion into the seat bottom assembly, attaching the hook and loop fasteners.

(7) Connect the upper end of the gas spring support to the bracket on the outboard side of the seat back.

(8) If removed, install the restraint system belt latch and stiffening sleeve assembly (See “Figure 4” on page 252015, Detail D) to the inboard (aisle) side of the seat pedestal.

C. Gas Spring Support

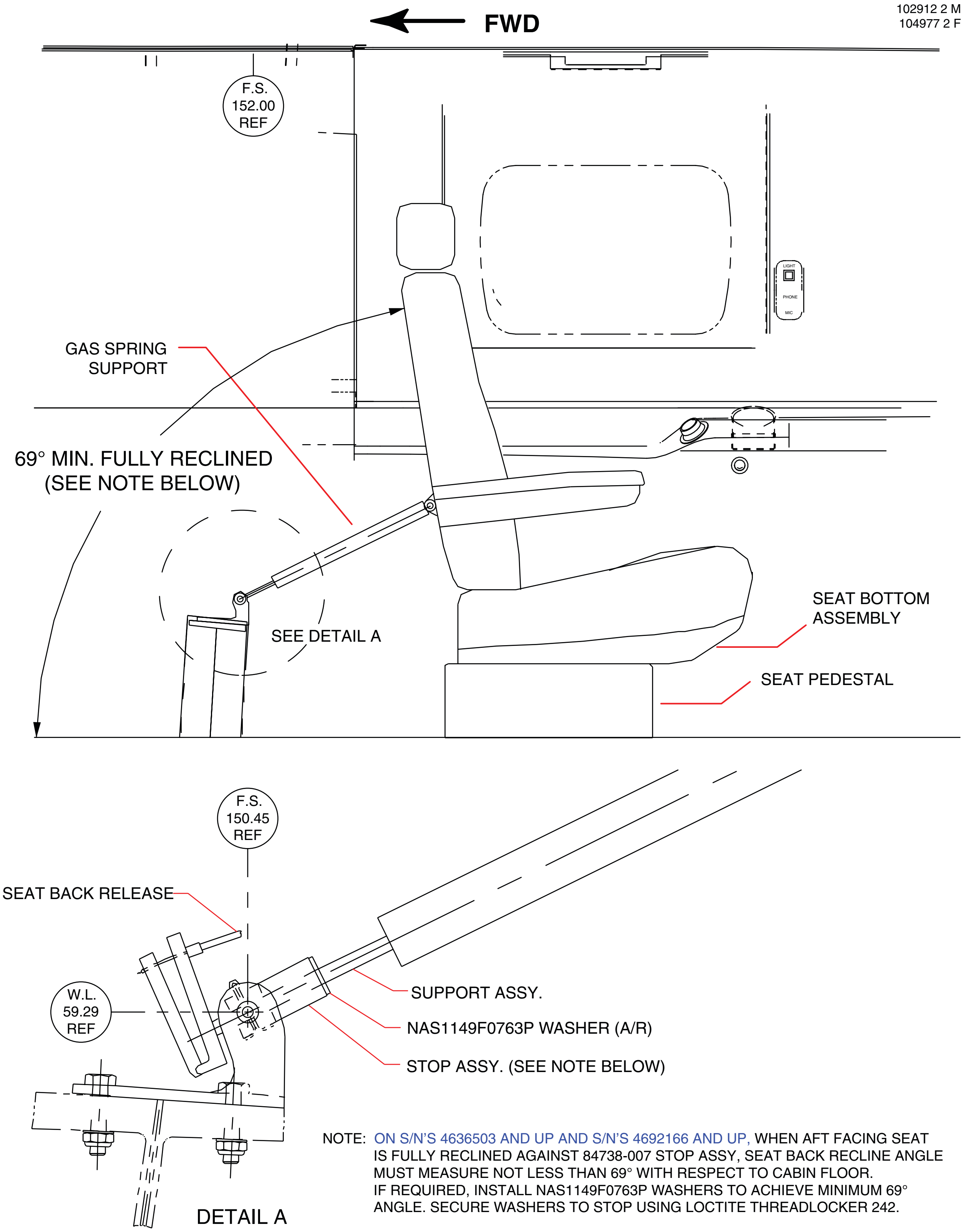
WARNING: DO NOT ATTEMPT TO DISASSEMBLE THE GAS SPRING.

(1) Removal

(a) Disconnect the upper end of the gas spring support from the bracket on the outboard side of the seat back.

(b) Disconnect the lower end of the gas spring support from its lower bracket by removing the cotter pins, washers (if any) and pins from the gas spring support’s attach stop.

(c) Remove the gas spring support.



NOTE: ON S/N'S 4636503 AND UP AND S/N'S 4692166 AND UP, WHEN AFT FACING SEAT IS FULLY RECLINED AGAINST 84738-007 STOP ASSY, SEAT BACK RECLINE ANGLE MUST MEASURE NOT LESS THAN 69° WITH RESPECT TO CABIN FLOOR. IF REQUIRED, INSTALL NAS1149F0763P WASHERS TO ACHIEVE MINIMUM 69° ANGLE. SECURE WASHERS TO STOP USING LOCTITE THREADLOCKER 242.

Passenger Seats - Aft Facing
Figure 1

(2) Installation

(a) Place the gas spring support in position.

(b) Connect the lower end of the gas spring to its lower bracket with the washers (if any), pins and cotter pins.

(c) Connect the upper end of the gas spring to its upper support (on the outboard side of the seat back) with the bolt, washer, and nut.

1) In S/N's 4636001 thru 4636502 and S/N's 4692001 thru 4692165, secure the nut with a cotter pin.

2) In S/N's 4636503 and up and S/N's 4692166 and up, ensure the nut is installed head outboard. Apply Loctite Threadlocker No. 242 and torque 50–60 in.-lbs.

(3) Adjustment

S/N's 4636503 and up and S/N's 4692166 and up only.

If gas spring support is replaced, add NAS1149F0763P washers between stop assembly and gas spring support cylinder to limit fully reclined seat back angle to no less than 69 degrees as shown in "Figure 1".

(a) Disconnect the lower end of the gas spring support from its lower bracket by removing cotter pins, washers (if any), and pins from the gas spring support stop assembly.

(b) Remove stop assembly from gas spring support rod.

(c) Add NAS1149F0763P washers as required over gas spring support rod.

NOTE: Start with the number of washers used previously. Indeed, these same washers may remain attached to the stop assembly.

(d) Replace stop assembly on end of gas spring support rod.

(e) Connect the lower end of the gas spring support to its lower bracket with cotter pins, washers (if any), and pins previously removed from the gas spring support stop assembly.

(f) Check fully reclined seat back angle per "Figure 1".

(g) Repeat steps (a) thru (f) until desired fully reclined seat back angle is achieved within the limit specified.

(h) When correct number of washers identified, ensure the washers are secured to the stop assembly with Loctite Threadlocker No. 242.

2. Forward Facing Passenger Seats

Not Shown.

A. Removal

(1) Remove the seat bottom filler assembly from the seat bottom pan assembly to gain access into the interior of the pan.

(2) Disconnect the cable from the forward end of the seat back actuator and remove the seat back actuator from the channel as follows:

(a) Remove the forward bolt from the channel and remove the seat back release cover from the channel.

(b) Remove the two cotter pins from the retaining pins in the stop assembly and remove the retaining pins from the stop assembly.

(c) Remove the nut, bolt and washers from the aft end of the seat back actuator. Lift the actuator from the channel.

(3) Remove the bolts that hold the channel, seat back supports and armrest supports to the bottom of the seat pan. Lift the seat back with its supports from the seat assembly.

(4) The armrest may be removed from the side of the pan assembly by removing the nuts, bolts and washers that attach the armrest to the armrest support/side of the pan assembly.

(5) Remove the pan assembly from the floor.

B. Installation

(1) Position the pan assembly on the floor.

(2) Attach the armrest (if previously removed) to the side of the pan with the nuts, bolts and washers.

(3) Secure the channel to the outboard side of the pan bottom/floor with the aft three bolts and washers.

(4) Secure the armrest support and the seat back assembly support to the inboard side of the pan bottom with the bolts and washers. Install the forward inboard bolt.

(5) Install the seat back actuator into the channel as follows:

(a) Position the seat back actuator into the channel.

(b) Slide the bolt into the aft holes in the channel, placing on the bolt (in order): washer (if required), bushing, aft fitting of seat back actuator, roller, washer (if required) and nut.

(c) Insert a pin in each side of the stop assembly and secure them with cotter pins.

(d) Secure the seat back release cover to the forward portion of the channel with the nut and bolt.

(e) Attach the cable assembly to the forward end of the seat back actuator.

(6) Attach the seat back/seat back support to the bottom of the pan assembly and aft portion of channel.

(7) Reinstall the seat bottom filler assembly on the seat bottom pan assembly.

3. Carpets

Carpets are individually fastened to the floor with hook & loop fasteners and adhesive (Hysol EA9309 NA).

4. Interior Panels - Cabin

A. General

The aircraft interior includes overhead, wall (side) and aft panels that serve aesthetic and functional purposes. The panels are composed of lightweight fiberglass and covered with soft fabric. Some panel assemblies include electrical terminals or other attachments.

Interior panels are interlocking and, accordingly, must be removed and installed in order. The general rules are:

(1) Remove them from the aft end of the cabin to the front of the cockpit. Remove headliners before side or window panels.

(2) Install them from the front of the cockpit to the aft end of the cabin. Install side or window panels before headliners.

NOTE: Use two people to handle the interior panels while removing and installing them, when practical. While not heavy, the panels can be unwieldy.

B. Removal

CAUTION: AS YOU REMOVE INTERIOR PANELS, DISCONNECT ANY ELECTRICAL TERMINALS, AIR VENTS, ETC. INCLUDED IN THE PANEL ASSEMBLIES.

NOTE: Each shoulder harness threads through an opening in a side or window panel, requiring that at least the lower outboard attachment of the restraint system must be separated from its attachment point and the belt buckle unlatched to remove an associated side or window panel.

(1) Baggage Closeout Panel

See “Figure 2” on page 25206 and “Figure 4” on page 252014.

The baggage closeout panel (aft cabin panel) is attached to the fuselage by hook and loop fasteners. Remove it as follows:

(a) If installed, detach the cargo net from the aft side panels.

(b) Remove the baggage closeout panel by pulling the two handles at its bottom, then steadily pull the panel free from the hook and loop fasteners along much of its edge.

(c) Remove it from the airplane.

(2) Cabin Headliner Panel

See “Figure 2” on page 25206 and “Figure 3” on page 252012.

(a) Remove the baggage closeout panel as described above.

(b) Remove the lighting fixture bezels. The bezels are attached over the headliner with push-pull, ball stud fasteners. Pull the bezels off to remove them.

(c) Remove the headliner by pulling down on it from the forward end to the aft end of the panel.

NOTE: It is not necessary to remove the light fixtures from the overhead.

(3) Aft (Baggage) Side Panels

Each panel is attached to the fuselage by hook and loop fasteners and attached by retainers to the lower aft side panel.

(a) Remove the baggage closeout panel as described above.

(b) Remove the cabin headliner panel as described above.

(c) Unlatch the seat belt buckle and disconnect the aft bench seat restraint system outboard lower end fitting per Restraint System - Cabin, “Removal” on page 252013.

(d) Remove the buttons and underlying screws for the shoulder harness bezel in the side panel assembly. Remove the plastic bezel and inner plate, and slip them off of the shoulder harness / seat belt.

(e) Remove the panel from the wall.

(f) Hold the panel securely and separate the safety belts from the panel, as follows:

1) Pull the belt assembly through the panel’s belt opening, toward the reel (i.e., from the finished side to the back side of the panel).

2) Carefully guide the metal buckle and end fittings through the opening.

3) Pull the belt through opening and, to secure it, drape the belt behind the bench seat.

(g) Remove the panel from the aircraft. Place it in a clean, secure area.

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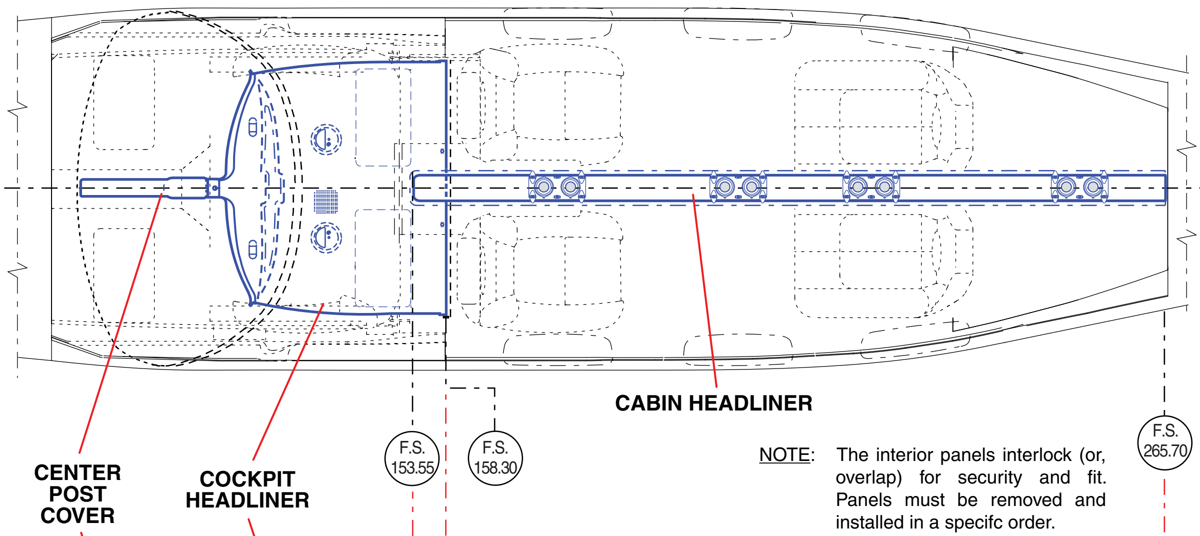
COCKPIT AND CABIN - OVERHEAD PANELS

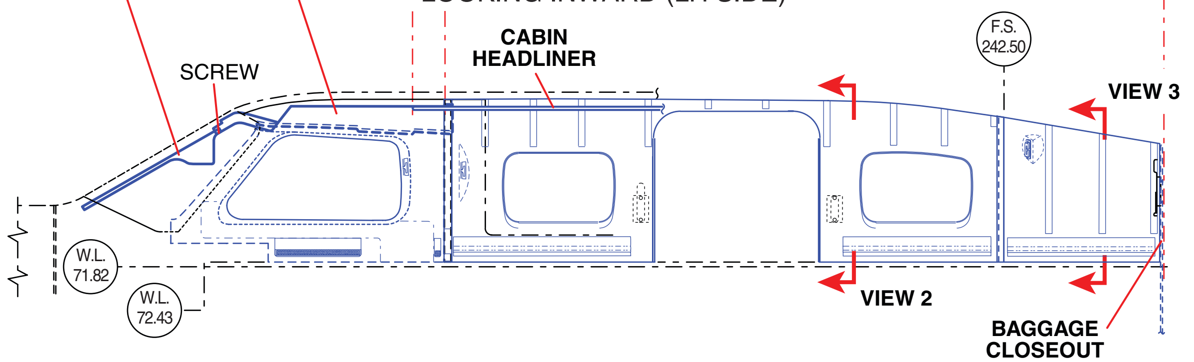
LOOKING DOWN



NOTE: The interior panels interlock (or, overlap) for security and fit. Panels must be removed and installed in a specifc order.

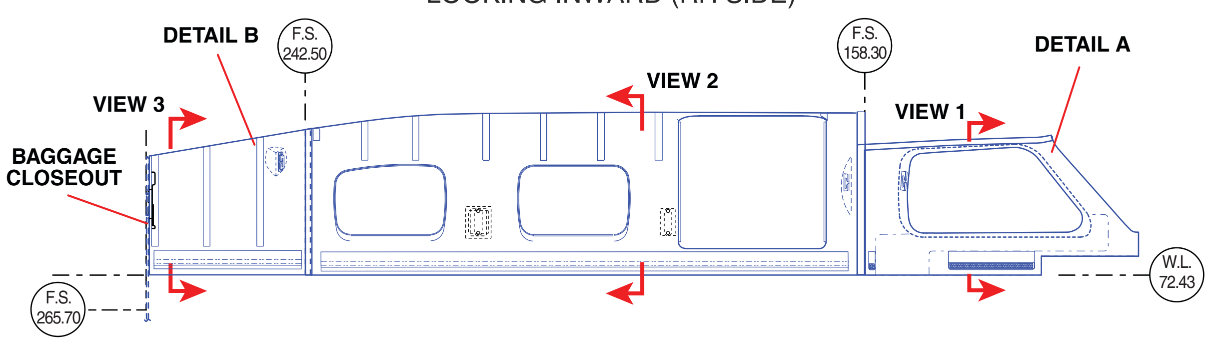
COCKPIT AND CABIN - OVERHEAD AND LH WINDOW PANELS

LOOKING INWARD (LH SIDE)

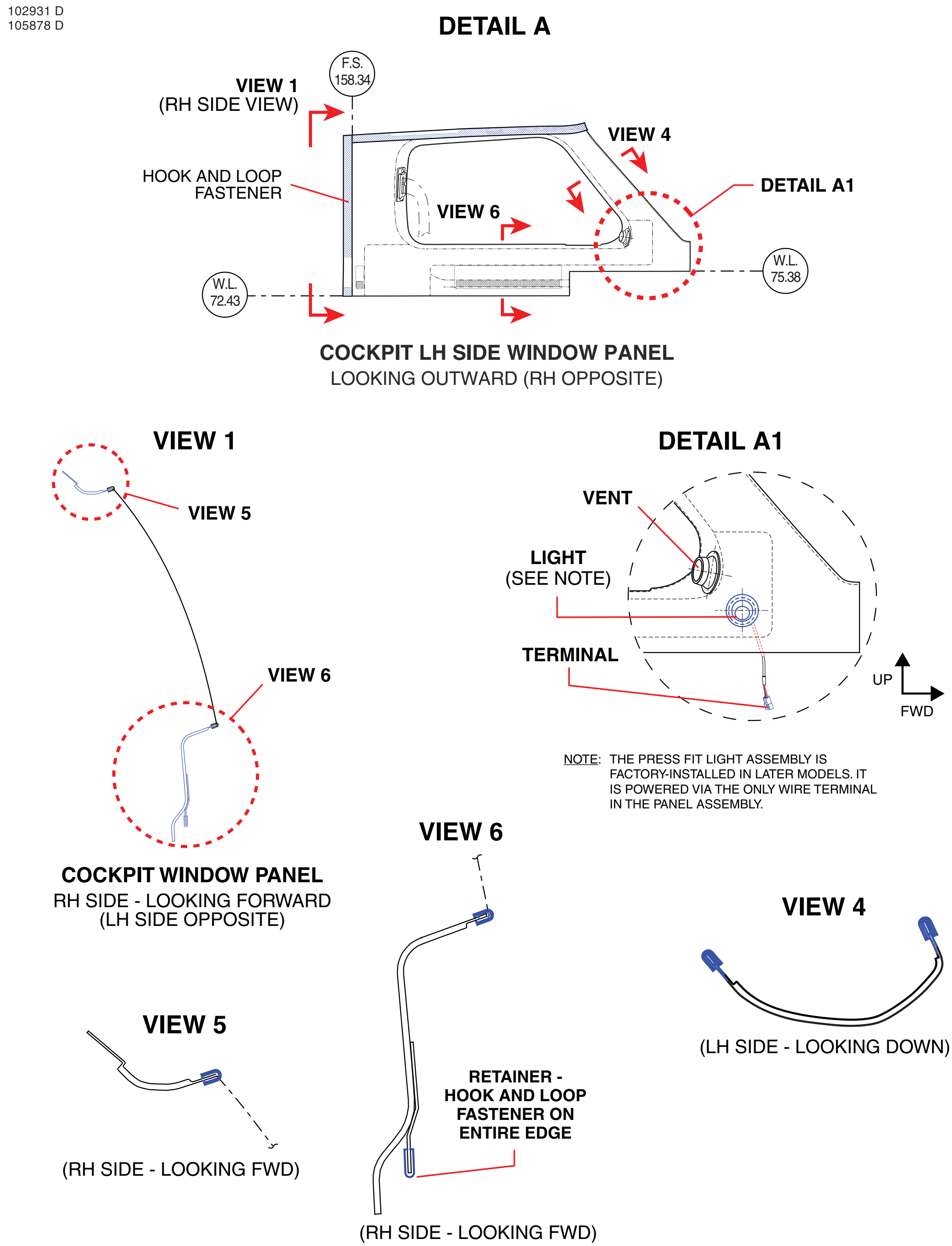


CABIN AND COCKPIT - RH WINDOW PANELS

LOOKING INWARD (RH SIDE)

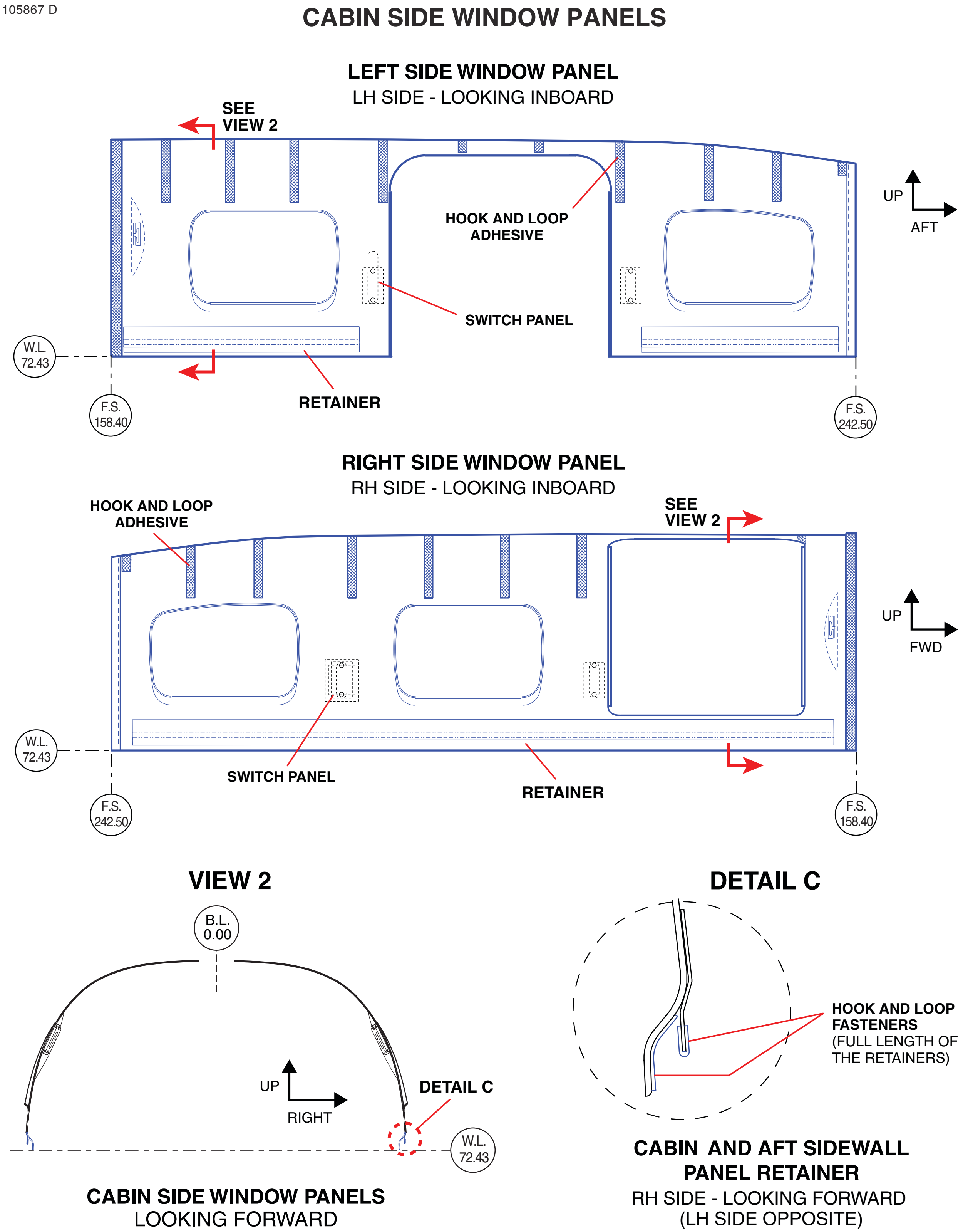


Cockpit and Cabin Interior Panels
Figure 2 (Sheet 1 of 4)



Cockpit and Cabin Interior Panels
Figure 2 (Sheet 2 of 4)

102918 D
105867 D

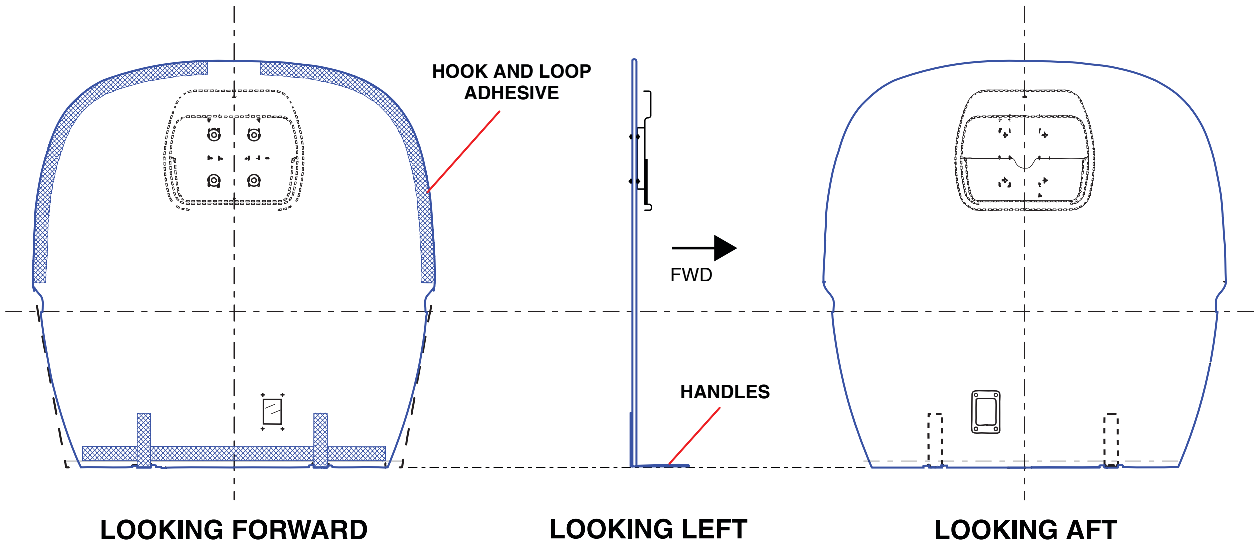


Cockpit and Cabin Interior Panels
Figure 2 (Sheet 3 of 4)

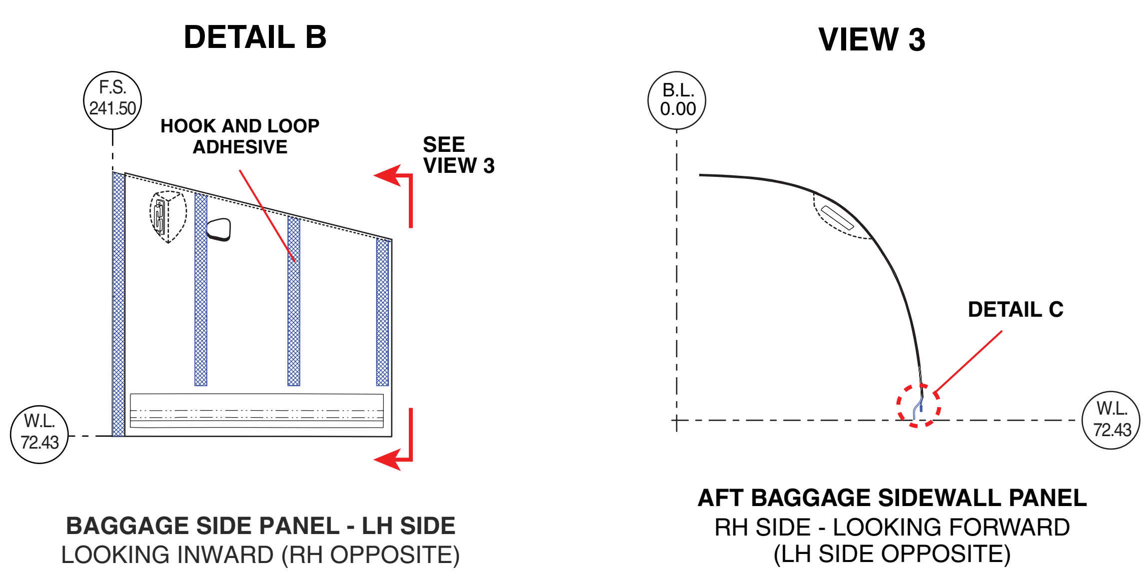
25208

102922 NEW
102923 F

BAGGAGE CLOSEOUT PANEL



AFT SIDEWALL PANEL



Cockpit and Cabin Interior Panels
Figure 2 (Sheet 4 of 4)

(4) Cabin Side Window Panels

Each panel is attached to the fuselage by hook and loop fasteners and attached by retainers to the lower cabin side panel. Remove the panels as follows:

(a) Remove the baggage closeout panel as described above.

(b) Remove the cabin headliner panel as described above.

(c) Remove the aft (baggage) side panel as described above.

(d) Unlatch the seat belt buckle and disconnect the aft-facing seat restraint system outboard lower end fitting per “Removal” on page 25201 under Aft Facing Passenger Seats.

(e) Remove the buttons and underlying screws for the seat belt bezel in the panel assembly. Remove the plastic bezel and inner plate, then slip them off of the shoulder harness / seat belt.

(f) Begin to detach the window panel.

(g) Carefully disconnect electrical terminals, affixed to a switch panel, on the back of the interior panel.

(h) Remove the panel from the wall. Hold the panel securely and separate the safety belt from the panel, as follows:

1) Pull the belt assembly through the panel’s belt opening, toward the reel (i.e., from the finished side to the back side of the panel).

2) Carefully guide the metal buckle and end fittings through the opening.

3) Pull the belt through the opening. Draw the belt assembly over the headrest of the crew seat back to secure it.

(i) Remove the panel from the aircraft. Place it in a clean, secure area.

C. Installation

CAUTION: AS YOU INSTALL INTERIOR PANELS, BE SURE TO RECONNECT ANY ELECTRICAL TERMINALS AND VENTS INCLUDED IN THE PANEL ASSEMBLIES.

NOTE: Each shoulder harness threads through an opening in a side or window panel. Ensure that the upper attachments and inertia reel plates are installed and the inertia reel properly adjusted per “Installation” on page 252016, before attempting to install the associated side or window panel.

(1) Cabin Side Window Panels

See “Figure 2” on page 25208, Sheet 3.

(a) Install cockpit interior panels per “Installation” on page 25108 under Interior Panels - Cockpit.

(b) Position the panel adjacent to its final installed location.

(c) Hold the panel securely.

(d) Thread the aft-facing seat restraint belt assembly through the belt opening. Thread it from the back side to the finished side of the panel:

1) Carefully guide the metal end fittings and buckle through the opening.

2) Wrap the belt loosely over the crew seat headrest for security and to allow access to the side panel for installation.

(e) Install the panel assembly:

1) Connect the electrical terminals for the two switch panels in the panel assembly.

2) Fit the retainers at the bottom of the panel (see “Figure 5” on page 251012, Detail C) to the lower side panel, being mindful of aligning the panel with the windows, door and fuselage.

3) Press the panel against the fuselage to join the hook and loop fasteners, being sure to align the panel shape to the windows and door.

(f) Thread the metal end fittings and buckle through the shoulder harness inner plate and bezel and slide them both up the shoulder harness to near their installed position.

(g) Attach the inner plate and screws to the interior panel with screws. Attach the plastic bezel to the inner plate with button fasteners.

(h) If desired, complete aft-facing seat installation per “Installation” on page 25201 under Aft-Facing Passenger Seats.

(2) Aft Side Panels

See “Figure 2” on page 25209, Sheet 4.

(a) Install cockpit interior panels per “Installation” on page 25108 under Interior Panels - Cockpit.

(b) Install cabin side window panel(s) as described above.

(c) Position the panel adjacent to its final installed location.

(d) Hold the panel securely.

(e) Thread the corresponding safety belt assemblies through the belt opening. Thread it from the back side to the finished side of the panel:

1) Carefully guide the metal end fittings and buckle through the belt opening in the panel.

2) Drape the belt behind the bench seat headrest for security and accessing the panel.

(f) Install the panel assembly:

1) Fit the retainers at the bottom of the panel to the lower aft panel, being mindful of fitting the panel to the fuselage.

2) Press the panel against the fuselage to join the hook and loop fasteners, being sure to align the panel with the fuselage.

(g) Thread the metal end fittings and buckle through the shoulder harness inner plate and bezel and slide them both up the shoulder harness to near their installed position.

(h) Attach the inner plate and screws to the interior panel with screws. Attach the plastic bezel to the inner plate with button fasteners.

(i) If desired, reattach aft bench seat restraint system lower end fitting attachments per Restraint System - Cabin, “Installation” on page 252016.

(3) Cabin Headliner

See “Figure 3”.

The cabin headliner assembly covers the four (4) overhead lighting assemblies. Over the headliner are bezels for the lighting. The headliners and bezels are attached to the support structure with ball stud fasteners.

NOTE: Prior to installing the headliner, install any overhead lighting fixtures that were removed and connect any electrical terminals that may have been detached.

(a) Begin to install the headliner by inserting the forward panel push-pull fasteners.

(b) The lighting bezels attach to the support structure over the cabin headliner panel with push-pull, ball stud fasteners. To complete the installation, push the bezels onto the headliner.

(4) Baggage Closeout Panel

See “Figure 2” on page 25209, Sheet 4

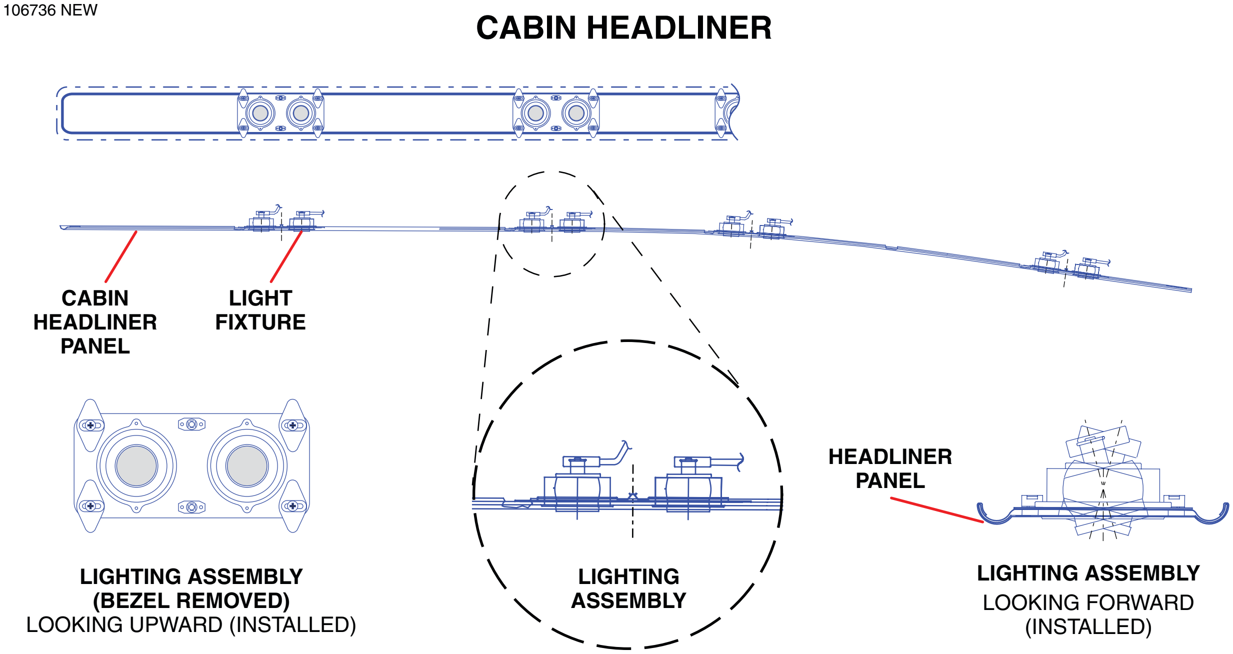
The baggage closeout panel is attached by hook and loop fasteners along its edges. The panel is positioned over the aft end (as installed) of the aft side panels.

(a) Set the bottom edge of the panel behind the baggage area floor structure. Be sure that the panel handles are laying forward of the panel.

(b) Press the panel edges against the support structure to connect the hook and loop fasteners.

(c) If included, attach the cargo net to the metal fittings on the aft side panels.

102901 B
106736 NEW



Cabin Headliner and Lighting Assembly
Figure 3

5. Restraint System - Cabin

An integrated shoulder harness / lap belt restraint system (“Figure 4” on page 251011) is installed for each crew seat (“Figure 5” on page 251012). They each include a lap belt, shoulder harness, inertia reel, buckle, clasp assembly, end fittings, and attachments. The restraint system upper attachments and inertia reels are located behind interior panels (see “Figure 5” on page 251012).

To remove, install or adjust the restraints, several lightweight, fabric-covered interior panels must be loosened or removed. The panels are interlocking and must be removed and installed in a specific order, see Interior Panels - Cabin, “General” on page 25204.

A. Inspection

Inspect per “Inspection” on page 251010 under Restraint System.

B. Removal

The restraint system upper attachments and inertia reels are located behind the side window or side panels (see “Figure 5” on page 251012). To service or remove the reels or upper attachments, you must remove the associated side window or side panel per Interior Panels - Cabin, “Removal” on page 25205. Likewise, accessing the lower end fittings requires some seat disassembly.

Remove the lower end fitting attachments first.

NOTE: Remove crew restraints per “Removal” on page 251014 under Restraint System.

(1) Lower End Fitting Attachments

(a) Aft-Facing Seats

See “Figure 4” on page 252014.

The aft-facing lower end fitting attachements bolt to the seat pedestals. Accordingly, their removal procedures are per “Removal” on page 25201, under Aft-Facing Passenger Seats.

(b) Forward-Facing Bench Seat

See “Figure 5” on page 251012.

The bench seat restraint systems are attached to gussets concealed behind the seat bottom cushion, in the center and the outboard edges of the seat support structure.

1) Unlatch belt buckles from the latches at center of the bench seat. Draw belt assemblies to the outboard sides of the bench seat and secure them behind the seat back.

2) Remove the seat bottom cushion. Pull up along the entire front of the cushion detaching the hook and loop fasteners, then draw cushion forward until it is fully removed. Place cushion in a clean location.

3) The two inboard end fittings are attached to a gusset (located at B.L. 0.00, attached to F.S. 233.87). Remove the bolt assembly. (See “Figure 5” on page 251013, View 5.)

4) The outboard end fittings are attached to gussets along the left and right outer sides of the seat assembly. (The gussets are attached to the fuselage at B.L. 19.75, left and right.) Remove the bolt assembly. (See “Figure 5” on page 251013, View 4.)

(2) Belt Assemblies and Cabin Side Window or Aft Side Panels

Remove the cabin side window and/or aft side panel(s) per “Removal” on page 25205 under Interior Panels - Cabin.

(3) Upper Attachments and Inertia Reels

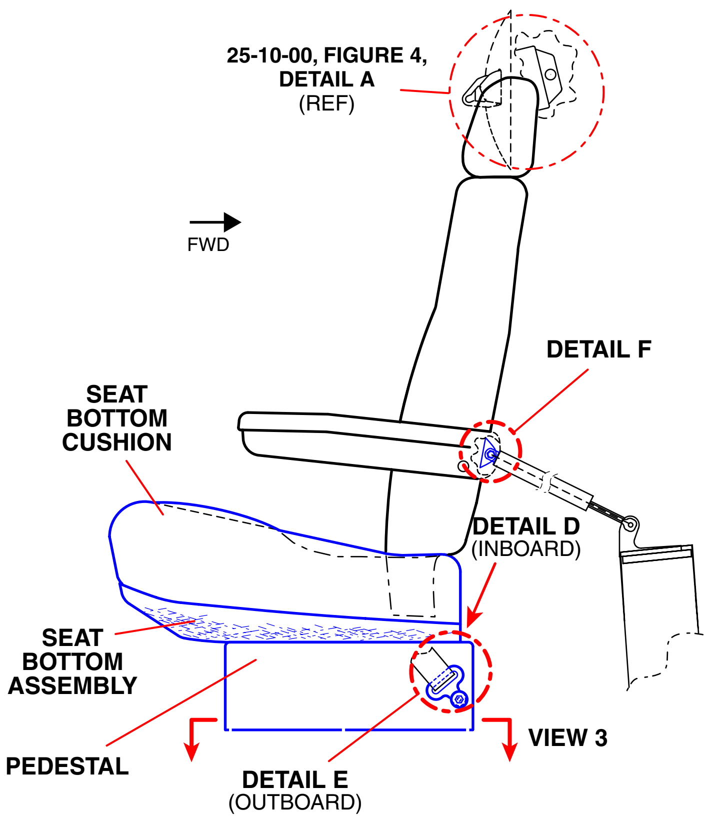
See “Figure 5” on page 251012.

The inertia reel is attached to a metal plate, which is in turn attached to a bracket. The bracket is attached to a fuselage support. (See 25-10-00, “Figure 5”, Sheet 1.)

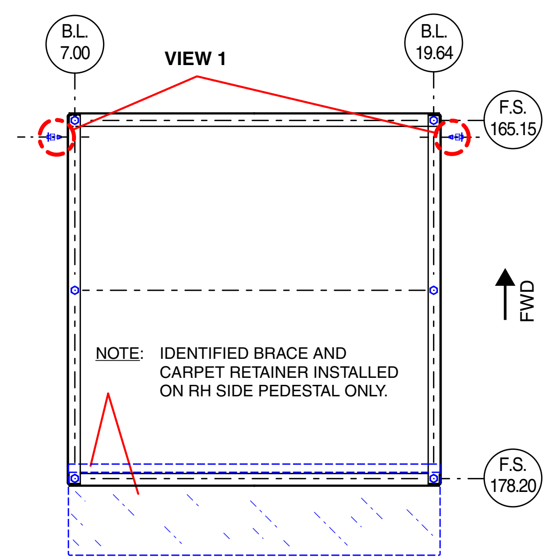
To remove the upper attachments and inertia reel plates, remove the bolt assembly from the support bracket.

102912 M

AFT-FACING SEAT - RH SIDE
LOOKING INBOARD
(LH OPPOSITE)



VIEW 3

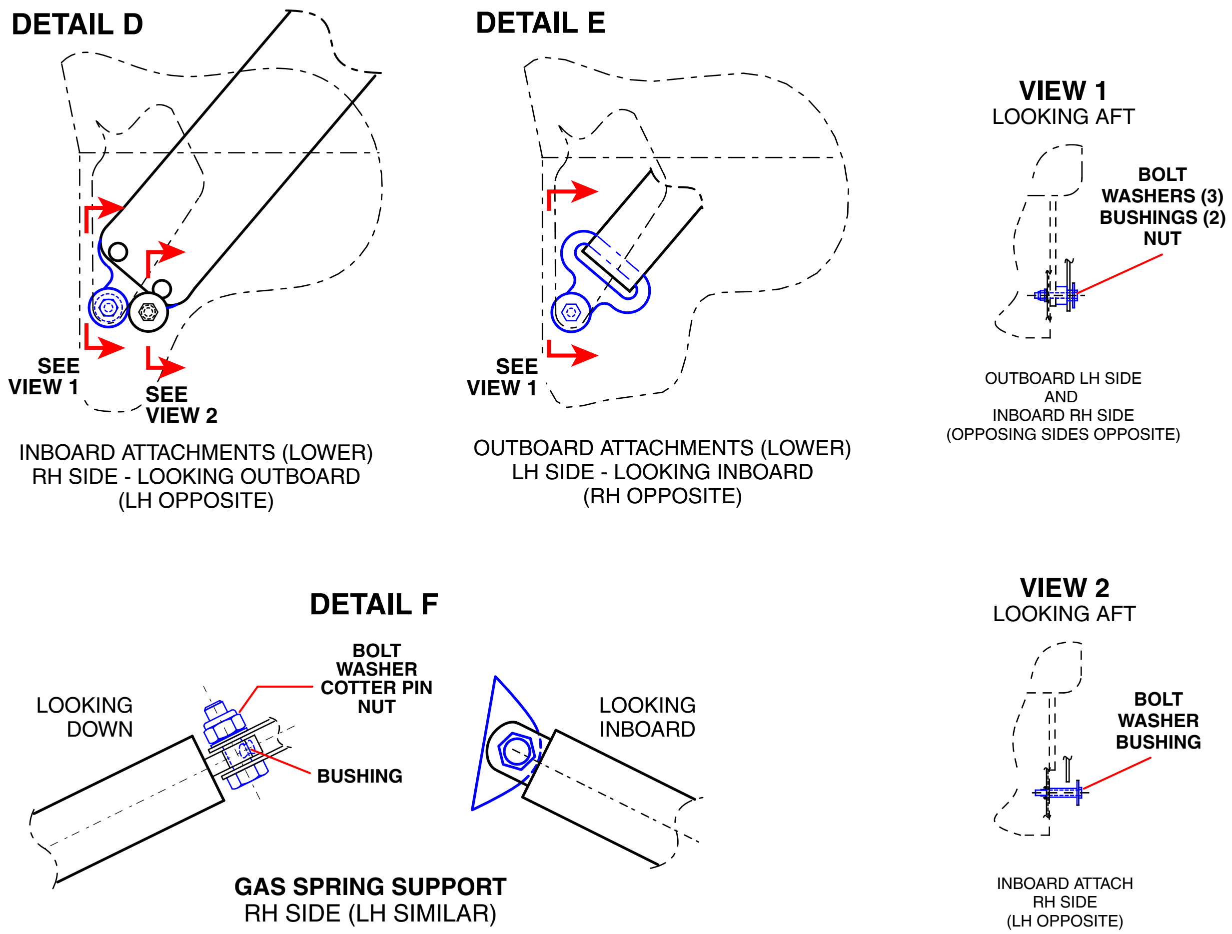


SEAT PEDESTAL
RH SIDE - LOOKING DOWN
(LH OPPOSITE - SEE NOTE)

NOTE: IN ORDER TO REMOVE THE OUTBOARD ATTACHMENTS, FOR THE AFT-FACING SEAT RESTRAINT SYSTEM, THE SEAT PEDESTAL MUST BE DETACHED. THE SEAT BOTTOM CUSHION AND THE SEAT CRADLE MUST FIRST BE REMOVED TO ACCESS THE BOLTS THAT ATTACH THE PEDESTAL TO THE FLOOR.

Aft-Facing Seat Restraints Installation
Figure 4 (Sheet 1 of 2)

252014



Aft-Facing Seat Restraints Installation
Figure 4 (Sheet 2 of 2)

C. Installation

NOTE: Install crew restraints per “Installation” on page 251014 under Restraint System.

As with removal, installation of the passenger restraint system necessarily involves interior panels and seats. To install the passenger integrated restraint system, proceed as follows:

(1) Upper Attachments and Inertia Reel Plates

See “Figure 4” on page 251011, and “Figure 5” on page 251012.

Install the restraint system upper attachments with bolt assemblies (see “Figure 5” on page 251012, Sheet 1) to bracket supports.

NOTE: As necessary, adjust the restraint system inertia reels before installing the interior panels. To adjust the inertia reels, see “Inertial Reel Adjustment” on page 251015 under Restraint System.

(2) Belt Assemblies and Cabin Side Window or Aft Side Panels

See “Figure 2” on page 25206.

Install the cabin side window and/or aft side panel(s) per Interior Panels - Cabin, “Installation” on page 252010, above.

(3) Lower End Fitting Attachments

See “Figure 5” on page 251012.

(a) Aft-Facing Seats

See “Figure 4” on page 252014.

1) If required, install and adjust the upper attachments and inertia Reels as described above.

2) If required, install the cabin side window panel per Interior Panels - Cabin, “Installation” on page 252010.

3) Contine per “Installation” on page 25201 under Aft-Facing Passenger Seats.

(b) Forward-Facing Bench Seat

See “Figure 5” on page 251012.

The bench seat restraint systems are attached to gussets concealed behind the seat bottom cushion, in the center and the outboard edges of the seat support structure.

1) Attach the restraint system outboard end fittings to the gussets at the left and right outer sides of the seat assembly (attached to the fuselage at B.L. 19.75, left and right). See “Figure 5” on page 251013, View 4.

2) Attach the two restraint system inboard end fittings to the center gusset (located at B.L. 0.00, attached to F.S. 233.87), with a bolt assembly.

See “Figure 5” on page 251013, View 5.

3) Install the seat bottom cushion. Fit the rear of the cushion under the bench back cushion. Push the front of the cushion downward to engage the hook and loop fasteners.

4) Insert the belt buckles into the latches at the center of the bench seat.

6. Cabinets (Optional)

The two cabinets are located between the crew seats and the forward passenger seats. They are secured to the floor by four bolts and washers (torque to 20-25 in.-lbs.).

7. Conference Table (Optional)

The conference table is located between the center and aft passenger seats on the right side panel just below window level opposite the cabin door. The table assembly is secured by three screws and washers.

CARGO COMPARTMENTS

1. Aft Baggage Net

The lower edge of the aft baggage net is secured to the F.S. 233.87 frame by four screws. The upper edge of the baggage net is secured to two stud assemblies (one on either side of the interior). The studs are located slightly aft of the shoulder harness inertia reel for the aft seat.

In the stowed position, the net may be rolled and left on the baggage floor at the base of the aft seat backs.

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EMERGENCY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This section contains information necessary to perform operational checks of the Emergency Locator Transmitter (ELT), with a pilot's remote switch. Included are the appropriate removal and installation instructions to facilitate battery replacement.

1. Ballast

CAUTION: DO NOT REMOVE BALLAST WEIGHT AND TRAY ASSEMBLY, IF INSTALLED.

A ballast weight and tray assembly may have been installed at the factory under the ELT mounting tray. If so, this is permanent ballast and must not be removed.

NOTE: Any requirement for ballast adjustment based on field modifications should be determined per lastest revision of AC 43.13-1. The structure for additional ballast installation(s) should be structurally substantiated based on FAA-approved data.

2. Artex ELT 110 Emergency Locator Transmitter (ELT)

This ELT was installed as standard equipment in PA-46-350P, S/N's 4636001 thru 4636425.

A. Description and Operation

The Artex ELT-110 transmits on 121.5 mHz and 243.0 mHz, and is designed to meet or exceed the requirements of TSO C91a and FAR Part 91. Electrical power for the ELT transmissions is totally supplied by its own self-contained battery. The battery must be replaced if the transmitter has been used in an emergency situation or if accumulated test time exceeds one hour, or no later than the replacement date marked on the transmitter label, whichever comes first.

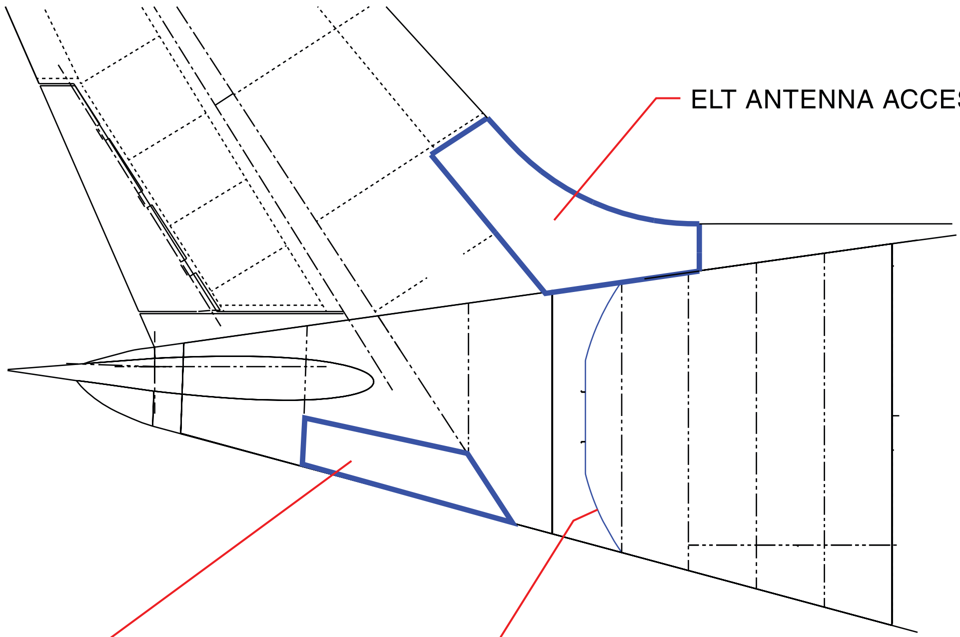
The Artex 110 cannot be accidentally activated by dropping the unit, handling it roughly, or during shipping. However, when properly mounted, and locked into its mounting tray, the ELT will activate in a crash, regardless of the cockpit remote switch and ELT switch position. The normal position of the ELT switch is in the down or OFF position. The normal position of the remote cockpit switch is in down or ARM position

Whenever the ELT is activated, a red light located just above the remote cockpit switch will blink to alert the pilot or maintenance personnel. Should the ELT be activated accidentally, it must be reset. To reset:

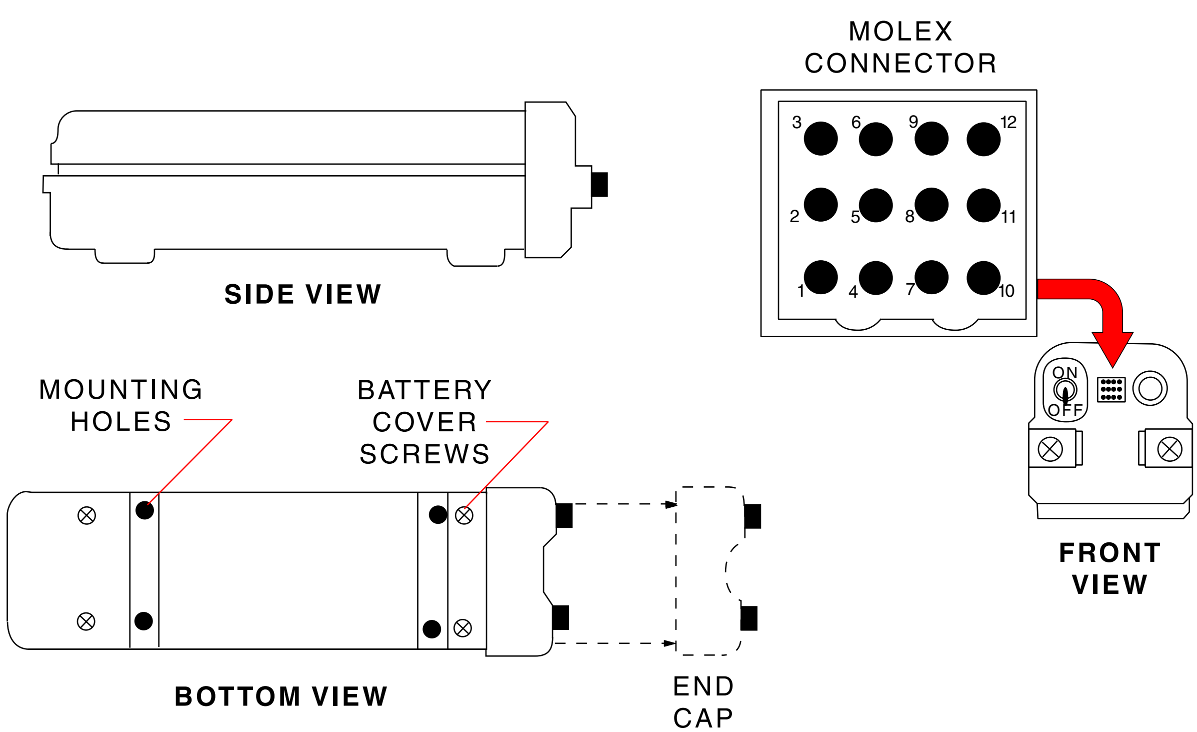
(1) Position the remote cockpit switch to ON, then immediately reposition it to ARM, or;

(2) Position the switch on the ELT to ON, then immediately reposition it to OFF.

The ELT is located in the tailcone and accessed through a large cover on the right lower side. The ELT antenna is underneath the dorsal fairing. (See "Figure 1" on page 25602.)



ELT Access Panels
Figure 1



Artex ELT 110-4
Figure 2

B. Inspections

(1) 100 Hour

Inspect per the 100 hour inspection (including the Antenna Test) under Artex ME406 ELT, below.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) Remove ELT from airplane per “Removal” below.

(b) Battery Inspection

1) Remove the four (4) securing screws from the bottom of the ELT.

2) Position the ELT product label (arrow) side down, and carefully lift the battery pack away from the ELT and lay it alongside the ELT.

3) Carefully disconnect the harness from the connector in the black plastic housing. Use a flat-bladed screwdriver to pry the connector out of its mating plug.

4) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

5) Inspect for broken wires, connections, or damage.

6) Ensure the battery housing is free of cracks or other visible damage.

7) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:

a) After use in an emergency;

b) After an inadvertent activation of unknown duration;

c) When the total of all known transmissions exceeds one (1) hour; or

d) On or before the battery replacement (expiration) date.

(c) Perform G-switch Check under Testing, below.

(d) Reinstall ELT into airplane per Installation, below.

(e) Perform Antenna Test per the procedure under Artex ME406 ELT, below.

(f) Reset ELT by turning ELT switch to “ON” then to “OFF/ARM” position.

(g) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

C. Removal

(1) Disconnect and remove ground (negative) battery cable from the airplane battery. Gain access as follows:

(a) Remove battery access plate from floor of forward baggage compartment.

(b) Pull back rubber battery terminal boot, loosen nuts and disconnect battery cable.

(2) At right lower side of tailcone, remove twenty-three (23) screws and washers and remove tailcone access panel.

(3) Remove ELT from the airplane by:

(a) Loosening the two screws on the front of the mounting tray and pull mounting tray cap off.

(b) Remove ELT Connections:

CAUTION: DO NOT USE CONTACT CLEANER ON ELT COMPONENTS. SUCH CHEMICAL AGENTS CAN BE HIGHLY DESTRUCTIVE TO THE MOUNTING HARDWARE AND ELT HOUSING.

1) Loosen the thumbscrews and remove the Molex and coax connectors.

2) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.

(c) Remove unit from airplane by lifting unit from the connector end. Carefull use of a flat-bladed screwdriver as a lever makes this step easier.

(d) Inspect the mounting tray for condition and security.

(e) Inspect all hardware to ensure the hardware is free of cracks or other obvious damage.

D. Installation

(1) Install unit into mounting tray:

(a) Connect molex and coax cables to ELT unit.

(b) Install mounting tray cap and secure to front of mounting tray with the two screws.

(2) Replace tailcone access panel and secure with twenty-three (23) screws and washers.

(3) Install ground (negative) battery cable to battery.

(4) Test transmitter per Testing below.

E. Battery Replacement

See “Figure 1” on page 25602 and “Figure 2” on page 25602

(1) Removal

(a) If required, remove the ELT from the airplane per Removal, above.

(b) Remove the four screws on the bottom of the ELT securing the battery pack.

(c) Disconnect battery pack connector from main unit. Use a flat-bladed screwdriver to pry the connector out of its mating plug.

(d) Remove battery pack from unit.

(2) Installation

(a) Securely plug in new battery pack connector to main unit.

(b) Immediately reset unit by positioning unit switch to ON, then to OFF.

(c) Fit new battery pack into place. Ensure all gaskets are properly aligned.

(d) Replace the four screws. Dress wires away from standoffs to avoid pinching wires between standoffs and the battery pack.

(e) Install ELT into airplane per Installation, above.

F. Testing

CAUTION: ALL ELT "ON" TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal.

The transmitter operates on the emergency frequencies of 121.5 and 243.0 MHz; both of these frequencies are monitored by the various FAA installations. Before performing any operational test of the ELT, the following precautions should be observed:

- Test should be no longer than three audio sweeps.
- Test should be conducted only within the time period made up of the first five minutes after any hour.
- If the operational tests must be made at a time not included within the first five minutes after the hour, the test should be coordinated with the closest FAA Tower or Flight Service Station.

(1) Installed Transmitter Test (Self-Test)

Every 90 Days

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune airplane communications receiver to 121.5 mHz and select SPKR on the audio panel.

(c) Turn the ELT aircraft panel switch to "ON", wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the "ARM" position.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to "ARM".

1) During the "ON" to "ARM" transition, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 and 8 on the tray connector.

2) If there is a problem, the processor will not immediately turn on the cockpit light during those first few sweeps. For example, if the jumper between pins 5 and 8 was not installed or was open, the cockpit light would flash momentarily upon ELT activation and the stay off for approximately three (3) seconds before beginning to flash continuously.

3) Repairs should be done only by a licensed aviation radio repair shop.

(2) Antenna Test (Each 100 Hours)

Use the procedure under Artex ME406 ELT, below.

(3) G-switch Check (Annually)

CAUTION: AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

A basic test of G-switch operation can be performed as follows:

(a) Setup

1) Remove the ELT from the airplane per Removal, above.

2) Fabricate a “shorting plug” by obtaining an appropriate connector and installing a jumper (i.e. - short) between Pins 5 and 8. (Artex sells a test plug for this purpose under part number 151-2012.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the elt activates without any pins shorted it is defective and should be returned for repair.

3) Install the “shorting plug” on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in Transmitter Test, below.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver, if desired).

3) Allow ELT to transmit only long enough to verify operation.

4) Reset ELT by turning ELT switch to “ON” then to ‘OFF” position.

G. Electrical Schematic

See 91-25-60.

3. Artex ME406 ELT

See "Figure 3" on page 25609.

A. Description and Operation

The Artex ME406 ELT is installed as standard equipment in PA-46-350P S/N's 4636426–4636715, 4636717–4636719 and up; and in PA-46R-350T S/N's 4692001 and up. It's a fully TSO-C126 / ETSO-2C126-certified, type AF (automatic fixed) beacon, 406 MHz ELT. All functions of the ME406 are under microprocessor control. A self-test routine checks ELT operation and installation, then presents the results as visual and auditory "error codes" to aid in troubleshooting and to indicate status. The ME406 is pre-programmed at the factory using a short message format. The following User Protocols are supported: Serial Number; Tail Number; 24-Bit Aircraft Address; and Aircraft Operator Designator/ Serial Number.

The battery pack consists of two D-size lithium cells mounted in a cover assembly, and is field-replaceable. Rated life is one hour of use or the expiration date on the battery pack, whichever comes first, as specified by FAR 91.207(c).

In the event of a crash, the ME406 activates automatically, and transmits the standard swept tone on 121.5 MHz lasting until battery power is gone. In addition, for the first 24 hours of operation, a 406 MHz signal is transmitting at 50 second intervals. This transmission lasts 440 ms and contains identification data programmed into the beacon and is received by Cospas-Sarsat satellites. Position accuracy of the 121.5 MHz signal is approximately 15-20 km radius about the transmitter. Accuracy of the 406 MHz signal is within about a 3 km radius.

In a crash, an acceleration-activated crash sensor (G-switch) turns the ELT "ON" automatically when the ELT experiences a change in velocity (or deceleration) of 4.5 fps ± 0.5 fps. Activation is also accomplished by means of the cockpit-mounted remote switch or the local switch on the ELT. To deactivate the ELT, set either switch to the "ON" position, then back to 'ARM'.

The ELT does not have an "OFF" position. Instead, a jumper between two pins on the front D-sub connector must be in place for the G-switch to activate the unit. The jumper is installed on the mating half of the connector so that when the connector is installed, the beacon is armed. This allows the beacon to be handled or shipped without "nuisance" activation (i.e. - front connector removed).

NOTE: The ELT can still be manually activated using the local switch on the front of the ELT. Care should be taken when transporting or shipping the ELT not to move the switch or allow packing material to become lodged such as to toggle the switch.

If the ELT is inadvertently activated, deactivate it by setting either switch to the "ON" position and then back to "ARM".

B. Troubleshooting

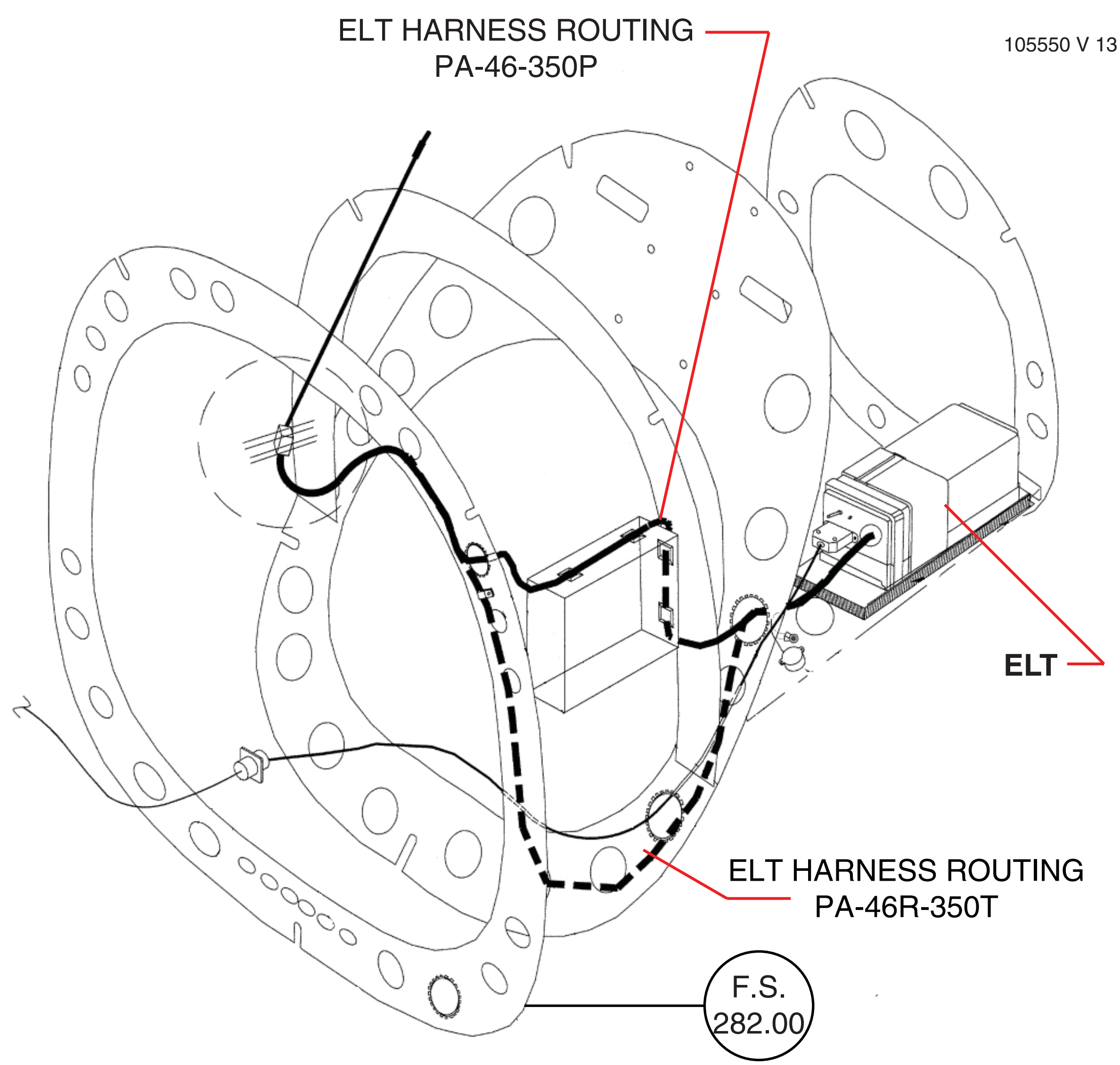
See "Chart 1" on page 25608.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING ME406 ELT

Trouble	Cause	Remedy
3 Flash Error after performing Self Test.	Bad load detect. Detects open or short condition on the antenna output or cable.	Check that the RF cable is connected and in good condition. Perform continuity check of center conductor and shield. Check for a shorted cable. Check for intermittent connection in the RF cable. If this error code persists there may be a problem with the antenna installation. Check this with a VSWR meter. Check the antenna for opens, shorts, resistive ground plane connection.
4 Flash Error after performing Self Test.	Low power detected. Occurs if output power is below about 33 dBm (2 watts) for the 406 signal or 17 dBm (50 mW) for the 121.5 MHz output. Also may indicate that 406 signal is off frequency.	Verify battery voltage. Replace battery if low voltage (~5.6 VDC) or if 7 Flash error is also present. Verify 406 MHz frequency. If bad, return for repair / replacement.
5 Flash Error after performinq Self Test.	Indicates that the ELT has not been proqrammed.	Read ELT 406 MHz signal to verify programming.
6 Flash Error after performing Self Test.	Indicates that G-switch loop between pins 5 and 12 at the D- sub connector is not installed. ELT will not activate during a crash.	Check that the harness D-sub jumper is installed iby verifying less thant one (1) ohm of resistance between pins 5 and 12. If missing, install jumper wire.
7 Flash Error after performing Self Test.	Indicates that the ELT battery has too much accumulated operation time (more than one hour) per regulation. May also indicate damage to the battery circuit.	Replace battery. If error does not clear after battery replacement, check continuity of battery circuit and correct function of circuit components.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING ME406 ELT

Trouble	Cause	Remedy
Remote Switch LED always on (steady).	Wiring error or frayed wires shorting out pins on back of Remote Switch.	Verify wiring. Verify integrity of all crimp or solder connections on harness.



ELT
RH SIDE LOOKING INBD

Artex ME406 ELT
Figure 3

C. Inspections

(1) 100 Hour

Each 100 hours time-in-service, inspect the ELT installation as follows:

(a) Inspect the ELT unit and mount for proper installation and secure mounting.

(b) Inspect wiring and conduits for proper routing, secure mounting, and obvious defects.

(c) Inspect bonding and shielding for proper installation and condition.

(d) Inspect antenna for condition, secure mounting, and proper operation per “Antenna Test” on page 256013.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) Remove ELT from airplane per ”D. Removal” on page 256011.

1) Inspect the mounting tray for condition and security.

2) Inspect all hardware to ensure the hardware is free of cracks or other obvious damage.

(b) Battery Inspection

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

1) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text: “BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

2) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

3) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

4) Inspect flex-circuit for broken connections or damage.

5) Ensure the battery housing is free of cracks or other visible damage.

6) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:

a) After use in an emergency;

b) When the transmitter has been in use for more than one (1) cumulative hour (i.e. - 7 flash error);

c) After an inadvertent activation of unknown duration; or

d) On or before the battery replacement (expiration) date.

(c) Perform “G-switch Check” on page 256013.

(d) Reinstall ELT into airplane per “Installation” on page 256011.

(e) Perform “Antenna Test” on page 256013.

(f) Perform “Digital Message Verification” on page 256014.

(g) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

(h) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(i) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

D. Removal

Refer to “Figure 1” on page 25602.

(1) Disconnect and remove ground (negative) battery cable from the airplane battery. Gain access as follows:

(a) Remove battery access plate from floor of forward baggage compartment.

(b) Pull back rubber battery terminal boot, loosen nuts and disconnect battery cable.

(2) At right lower side of tailcone, remove twenty-three (23) screws and washers and remove tailcone access panel.

(3) Remove ELT Connections

(a) Loosen the thumbscrews and remove the D-sub and RF connectors.

(b) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.

(4) Loosen the velcro strap and remove the ELT from its mounting tray.

E. Installation

(1) Insert the ELT into the mounting tray at an angle so that the locking ears at the end fit into the mounting tray locking slots.

(2) Fasten the Velcro strap around the ELT so that it is firmly held in place.

(3) Insert the D-sub and RF connectors ensuring that they are seated properly.

(4) Tighten the thumbscrews.

(5) At right lower side of tailcone, place tailcone access panel in position and secure with twenty-three (23) screws and washers.

(6) In the forward baggage compartment, connect ground (negative) battery cable to battery.

(7) Secure forward baggage compartment.

F. Battery Pack Replacement

NOTE: When replacing the battery pack, a replacement kit containing the battery pack, replacement gasket, hardware and labels is available.

(1) Remove ELT per Removal, above.

(2) Remove the battery pack as follows:

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

(a) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text:
“BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

(b) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

(c) Discard/recycle the old battery pack.

(3) Lay the new battery pack on the work surface with the batteries facing up.

(4) Install a replacement seal in the slot along the perimeter of the housing.

(5) Leaving the battery as it is, position the ELT over the battery pack with one hand and plug the flex-cable connector into the battery assembly using the other. The cable should not be twisted and the connector should ‘click’ into place.

NOTE: The battery connector is keyed to prevent incorrect installation.

(6) Mate the ELT to the battery, making sure that the seal is positioned correctly during the process.

(7) Replace the eight (8) securing screws and torque to 10 - 12 inch-lbs.

(8) Install ELT per “Installation” on page 256011.

(9) Perform “Installed Transmitter Test (Self Test)”, below.

(10) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.

G. Testing

CAUTION: ALL ELT “ON” TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal. If the unit operates for approximately 50 seconds, a 406 MHz distress signal is transmitted and is considered valid by the satellite system.

(1) Installed Transmitter Test (Self Test)

Every 90 Days

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune a receiver (usually the aircraft radio) to 121.5 MHz.

(c) Turn the ELT aircraft panel switch to “ON”, wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the “ARM” position while paying special attention of the LED activity upon entering the ‘ARM” condition.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to “ARM”.

1) During the “ON” to “ARM” transition, the microprocessor in the ELT checks the “G-Switch” (automatic activation switch) latching circuit, pins 5 & 12 on the D-sub connector at the ELT; the 406 MHz transmitter for proper RF output and a battery check.

2) If the ELT is working properly, the sequence following entry to the “ARM” condition will result in the panel LED staying illuminated for approximately one (1) second (one pulse), and then extinguishing. The buzzer should also sound once.

(e) If the panel LED and buzzer present more than one (1) pulse when switched to “ARM”, determine the problem from the list in “Chart 1” on page 25608.

(2) Antenna Test (Each 100 Hours)

Use a low quality AM broadcast receiver to determine if energy is being transmitted from the antenna.

(a) Hold the antenna of the radio (tuning dial on any setting) about six (6) inches from the activated ELT antenna. The ELT aural tone should be heard on the AM broadcast receiver.

(b) This is not a measured check, but it does provide confidence that the antenna is radiating sufficient power to aid search and rescue.

NOTE: Use of the aircraft's VHF receiver, tuned to 121.5 MHz, is not recommended. This receiver is more sensitive and could pick up a weak signal even if the radiating ELT antenna is disconnected. While it will confirm the ELT is active, it does not check the integrity of the ELT system

(3) G-switch Check (Annually)

A basic test of G-switch operation can be performed as follows:

CAUTION: EVEN WITHOUT AN ANTENNA CONNECTED, THE SIGNAL CAN BE RECEIVED BY A SATELLITE. AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

(a) Setup

1) Remove the ELT from the airplane per Removal, above.

2) Fabricate a "shorting plug" by obtaining a standard 15-pin D-sub connector and installing a jumper (i.e. - short) between Pins 5 and 12. Artex sells a test plug for this purpose under part number 150-1130.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the "shorting plug" on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in Transmitter Test, below.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver, if desired).

3) Allow ELT to transmit only long enough to verify operation.

4) Reset ELT by turning ELT switch to "ON" then to 'ARM" position.

(4) Digital Message Verification (Annually)

The ARTEX ME 406 MHz ELT transmits a 406 MHz message upon reset, which is encoded such that it will be ignored by the SAR satellite system. The 15-digit number contained in this transmission is used to register the ELT with the appropriate 406 MHz ELT registration authority. In the US, the National Oceanic and Atmospheric Administration (NOAA) maintains the database of registered ELTs. The information in this database provides the Search and Rescue system with aircraft identification data in the event an actual distress signal is transmitted.

Verify the 406 MHz digital message as follows:

(a) Required Equipment

A test set capable of receiving and decoding the message - i.e. - Artex ELT Test Set P/N 453-1000, or equivilent.

(b) Procedure

1) At right side of empennage, remove four (4) screws and remove ELT access panel.
2) Disconnect the antenna coax cable at the ELT, connect test set or terminate as applicable.
3) Perform all necessary steps to prepare Test Set to receive 406 MHz signal including (but not limited to) turning on power, activating program or any other steps required for the particular Test Set being used.
4) Perform the “Installed Transmitter Test (Self Test)” on page 256012.
5) Watch the screen on the Test Set to ensure that a message has been received. Repeat “Self Test” if necessary.
6) View message, ensure that all applicable information is correct (country code, aircraft ID, etc.).
7) The 15 digit ID hex ID (for example “ADC6492640D3411F1’) should match what is shown on the ELT product label. This is the 15 digit hex ID (Unique Identification Number or ‘UIN”) that is used to register the ELT.
8) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

H. Electrical Schematic

See 91-25-60.

4. Artex ELT 1000

Installed as standard equipment in PA-46-350P S/N's 4636716, 4636720 and up.

A. Description and Operation

The Artex ELT 1000 is a TSO-C126 / ETSO-2C126 certified and COSPAS-SARSAT TAC 251 approved, type AF (Automatic Fixed) beacon, which transmits on 121.5 and 406 MHz. When the 406 MHz signal is detected by the Cospas–Sarsat satellite LEO (Low Earth Orbit system), a position is calculated and the 121.5 MHz signal is used to home in on the crash site. The ELT 1000 is pre-programmed at the factory to use a transmitted short message format. The information contained in the message includes: (a) beacon serial number, (b) an aircraft identification or registration number, (c) a country of registration and country code and (d) position coordinates, if beacon is programmed to receive position data from the aircraft navigation system. ELT operation checks are performed by a series of self-tests with results as visual and auditory “error codes” to aid in troubleshooting and to indicate status.

The battery pack consists of two D-size lithium cells mounted in a covered assembly, and is field replaceable. Rated life is six (6) years unused or with total of transmissions exceeding one hour of use, whichever comes first, as specified by FAR 91.207(c). The lithium battery packs used on the Artex ELT 1000 are certified under TSO C142a.

In the event of a crash, the ELT 1000 activates automatically, and transmits the standard swept tone (homing signal) on 121.5 MHz lasting until battery power is gone, which with lithium cells is at least 50 hours. In addition, for the first 24 hours of operation, a 406 MHz signal is transmitting at 50 second intervals. This transmission lasts 440 ms and contains identification data programmed into the beacon and is received by Cospas-Sarsat satellites. Position accuracy of the 121.5 MHz signal is approximately 15–20 km radius about the transmitter. Accuracy of the 406 MHz signal is within about a 3 km radius. If position information is extracted from the aircraft navigation system, the accuracy improves to approximately 100 meters.

Artex ELT 1000 Emergency Locator Transmitter
Figure 4

In a crash, an acceleration activated crash sensor (G-switch) turns the ELT "ON" automatically when the ELT experiences a change in velocity (or deceleration) of 4.5 fps ± 0.5 fps. Activation is also accomplished by means of the cockpit mounted remote switch or the local switch on the ELT. To deactivate the ELT, set either switch to the "ON" position, then back to "ARM".

The ELT does not have an "OFF" position. Instead, a jumper between two pins on the front D-sub connector must be in place for the G-switch to activate the unit. The jumper is installed on the mating half of the connector so that when the connector is installed, the beacon is armed. When this connector is removed the ELT can not be activated. This allows the beacon to be handled or shipped without "nuisance" activation (i.e., front jumper connector removed).

NOTE: The ELT can still be manually activated using the local switch on the front of the ELT. Care should be taken when transporting or shipping the ELT not to move the switch or allow packing material to become lodged such as to toggle the switch.

If the ELT is inadvertently activated, deactivate it by setting either switch to the "ON" position and then back to "ARM". If the switch is already in the "ON" position, move it to the "ARM" position.

B. Troubleshooting

See "Chart 2".

C. Inspections

(1) 100 Hour

Each 100 hours time-in-service, inspect the ELT installation as follows:

(a) Inspect the ELT unit and mount for proper installation and secure mounting.

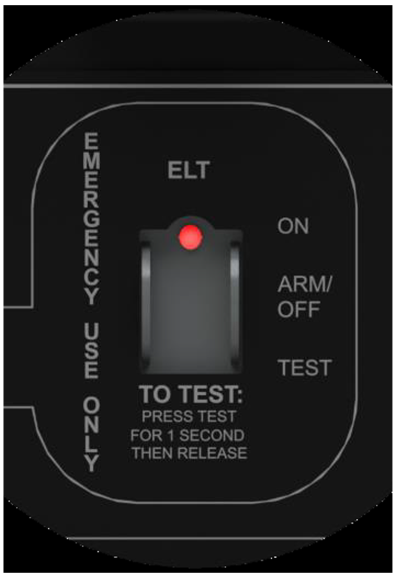
(b) Inspect wiring and conduits for proper routing, secure mounting, and obvious defects.

(c) Inspect bonding and shielding for proper installation and condition.

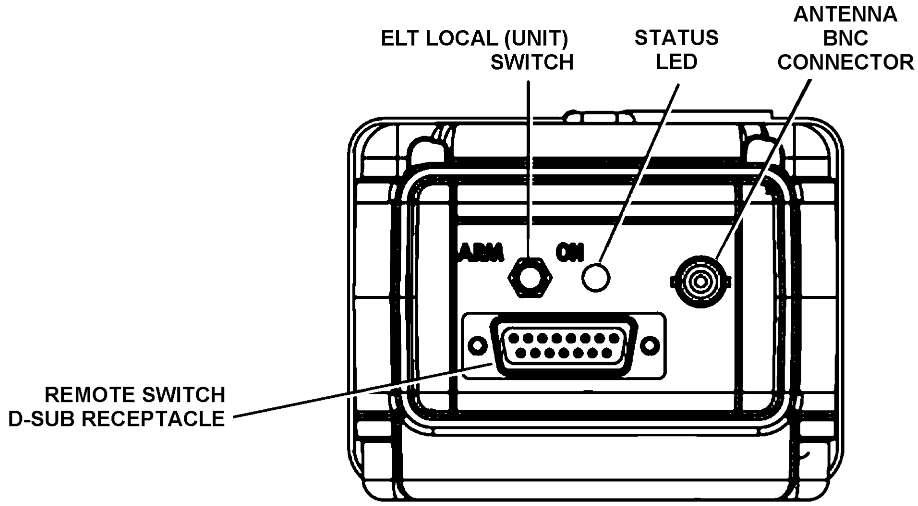
(d) Inspect antenna for condition, secure mounting, and proper operation (per "Antenna Test" on page 256023).

CHART 2
TROUBLESHOOTING ELT 1000

Trouble	Cause	Remedy
Remote switch LED always on (Steady).	Improper wiring.	Verify wiring.
	Short circuit.	Verify integrity of all crimp and solder connections.
ELT Will Not Turn Off.	**Turn ELT "OFF" Using Local Switch. If ELT turns OFF, then:**	
	Defective remote switch harness reset circuit wiring.	Check wiring continuity and repair as necessary.
	Defective remote switch.	Replace remote switch.
	If ELT Does Not Turn "OFF" Using Local Switch. then:	
	ELT defective.	Remove battery pack to disable ELT and return ELT to manufacturer for servicing.
Flash Errors.	Self-Test.	See "Chart 3" on page 256025.



COCKPIT (REMOTE)
SWITCH



ELT 1000 Manual Activation Switches
Figure 5

(e) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(f) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) Remove ELT from its mounting tray per “Removal”.

Inspect the mounting hardware. Ensure the hardware is free of cracks or other obvious damage.

(b) Battery Inspection

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

1) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text: “BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

2) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

3) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

4) Inspect flex-circuit for broken connections or damage.

5) Ensure the battery housing is free of cracks or other visible damage.

6) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:

a) After use in an emergency;

b) When the transmitter has been in use for more than one (1) cumulative hour (i.e., 7 flash error);

c) After an inadvertent activation of unknown duration; or

d) On or before the battery replacement (expiration) date.

(c) Perform “G-switch Check” on page 256024.

(d) Reinstall ELT into airplane per ”Installation”.

(e) Perform “Antenna Test” on page 256023.

NOTE: For a fee, http://www.406Test.com allows testing of the ELT through-the-satellite.

(f) Perform “Digital Message Verification” on page 256027.

(g) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

(h) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(i) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

D. Removal

Refer to “Figure 1” on page 25602, “Figure 6” and “Figure 7” on page 256020.

(1) At right lower side of tailcone, remove twenty-three (23) screws and washers and remove tailcone access panel.

(2) Remove ELT Connections

  (a) Loosen the thumbscrews and remove the D-sub and RF connectors.

  (b) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.

(3) Loosen the retention strap and remove the ELT from its mounting tray.

E. Installation

Refer to “Figure 1” on page 25602, “Figure 6” and “Figure 7” on page 256020.

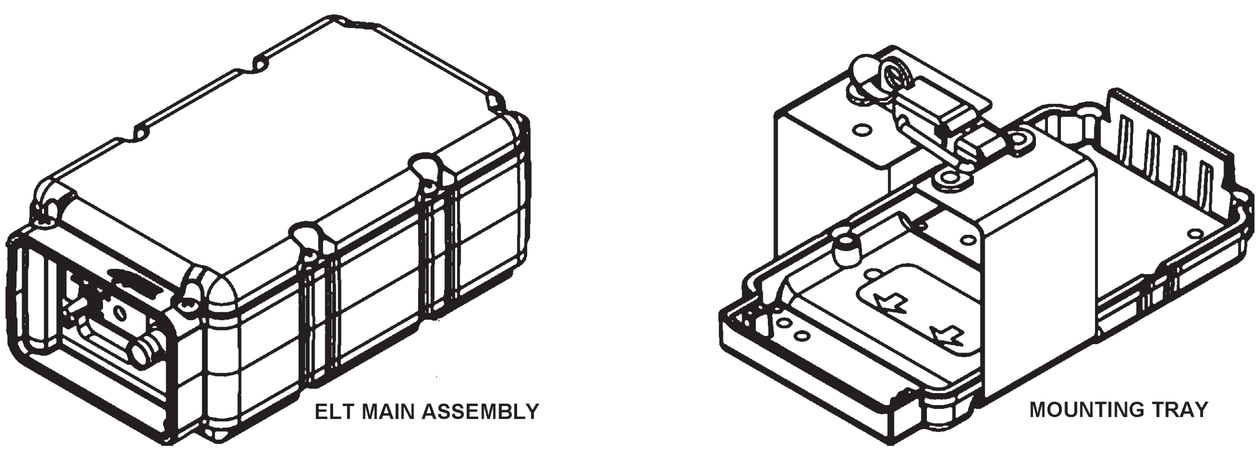
(1) Insert the ELT into the mounting tray at an angle so that the locking ears at the end fit into the mounting tray locking slots.

(2) Fasten the retention strap around the ELT so that it is firmly held in place.

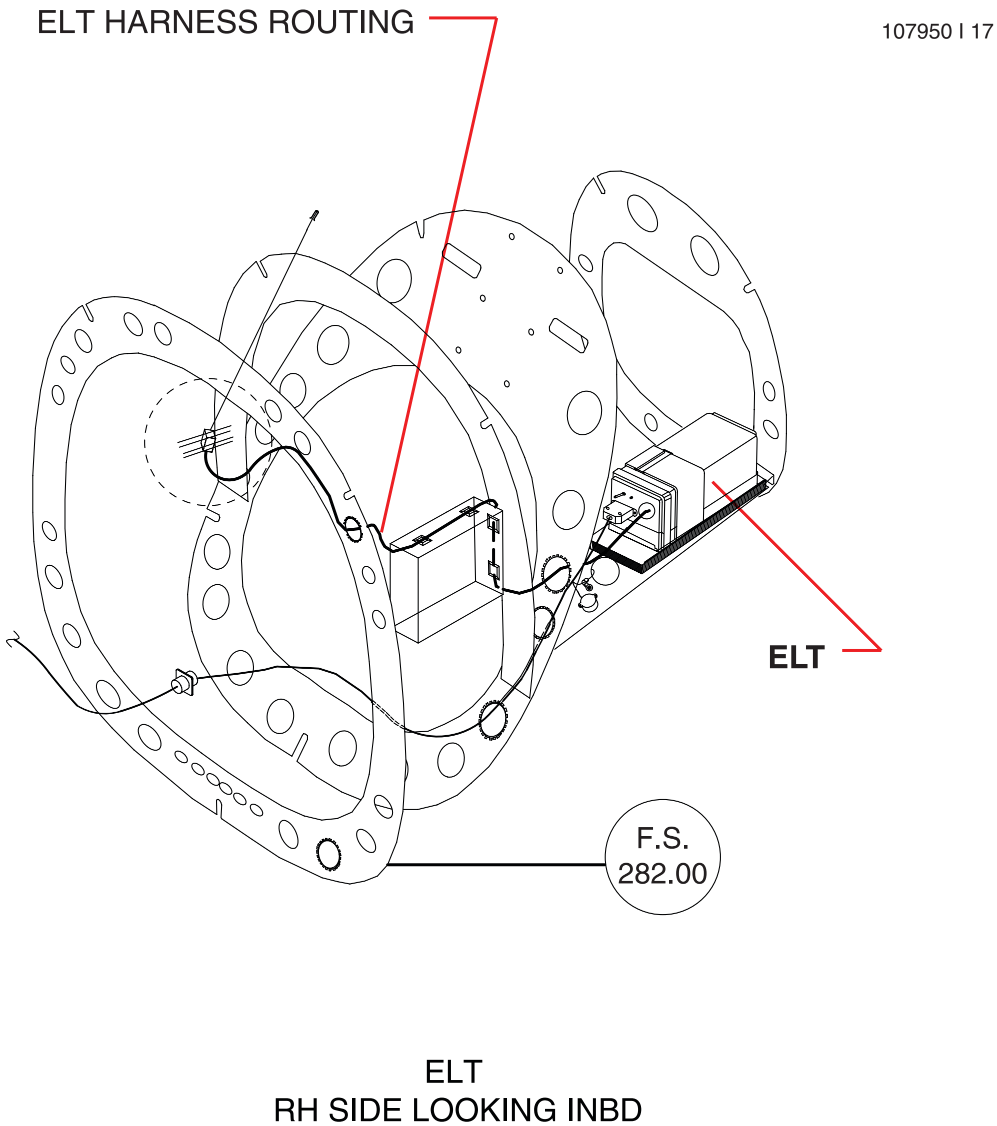
(3) Insert the D-sub and RF connectors ensuring that they are seated properly.

(4) Tighten the thumbscrews.

(5) At right lower side of tailcone, place tailcone access panel in position and secure with twenty-three (23) screws and washers.



ELT 1000 Assembly
Figure 6



Artex ELT 1000
Figure 7

(6) Installation Verification

NOTE: Perform the following after ELT installation is complete in the aircraft.

(a) Perform the following functional check within the first 5 minutes after the hour (UTC) as required by AC 43-13-1, Chapter 12 § 12–22, Note 3.

(b) Tune an AM receiver to 121.50 MHz and locate it approximately 20 feet to 30 feet from the ELT antenna.

(c) Initiate the self-test by momentarily pressing the cockpit remote switch into the TEST position for approximately 1 second.

(d) On the AM receiver ,listen for 3 audible sweeps, which takes about 1 second.

(e) Note the LED activity on the cockpit remote switch or the audio beeps from the ELT. If the ELT is working properly, the LED will stay on for approximately 1 second and then turn OFF accompanied

F. Battery Replacement

See “Figure 8” on page 256022.

NOTE: When replacing the battery pack, a replacement kit containing the battery pack, replacement gasket, hardware and labels is available.

(1) Remove ELT per “Removal” on page 256019.

(2) Remove the battery pack as follows:

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

(a) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text:
“BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

(b) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

(c) Discard/recycle the old battery pack.

(3) Lay the new battery pack on the work surface with the batteries facing up.

(4) Install a replacement seal in the slot along the perimeter of the housing.

(5) Leaving the battery as it is, position the ELT over the battery pack with one hand and plug the flex-cable connector into the battery assembly using the other. The cable should not be twisted and the connector should ‘click’ into place.

NOTE: The battery connector is keyed to prevent incorrect installation.

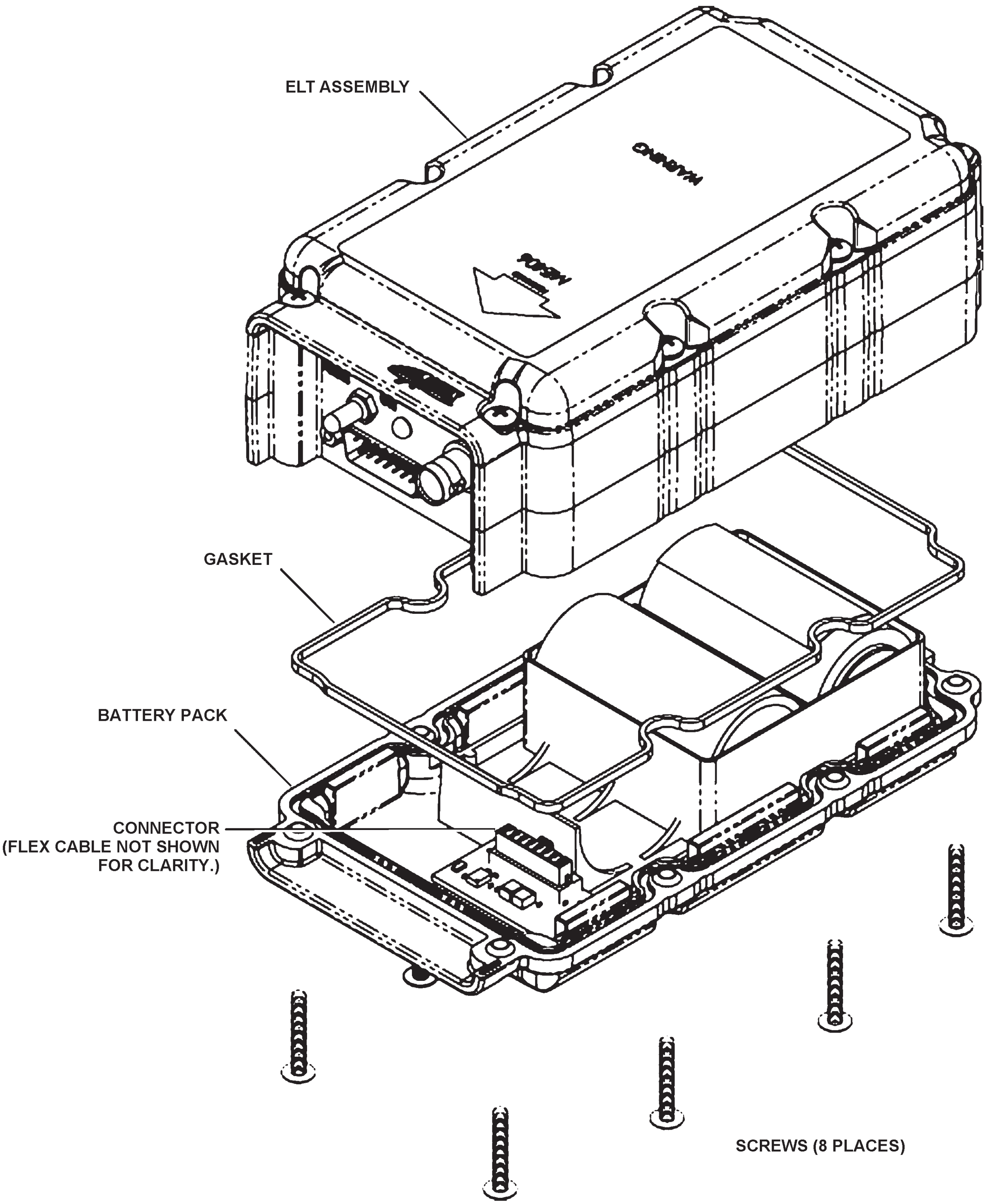
(6) Mate the ELT to the battery, making sure that the seal is positioned correctly during the process.

(7) Replace the eight (8) securing screws and torque to 10 - 12 inch-lbs.

(8) Install ELT per “Installation” on page 256019.

(9) Perform “Installed Transmitter Test (Self Test)” on page 256023.

(10) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.



Artex ELT 1000 Battery Assembly
Figure 8

G. Testing

CAUTION: ALL ELT "ON" TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal. If the unit operates for approximately 50 seconds, a 406 MHz distress signal is transmitted and is considered valid by the satellite system.

(1) Installed Transmitter Test (Self Test)

Every 90 Days

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune a receiver (usually the aircraft radio) to 121.5 MHz.

(c) Turn the ELT aircraft panel switch to "ON", wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the "ARM" position while paying special attention of the LED activity upon entering the 'ARM" condition.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to "ARM".

1) During the "ON" to "ARM" transition, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 & 12 on the D-sub connector at the ELT; the 406 MHz transmitter for proper RF output and a battery check.

2) If the ELT is working properly, the sequence following entry to the "ARM" condition will result in the panel LED staying illuminated for approximately one (1) second (one pulse), and then extinguishing. The buzzer should also sound once.

(e) If the panel LED and buzzer present more than one (1) pulse when switched to "ARM", determine the problem from the list in Chart 3.

(2) Antenna Test

Each 100 Hours use a low quality AM broadcast receiver to determine if energy is being transmitted from the antenna.

(a) Hold the antenna of the radio (tuning dial on any setting) about six (6) inches from the activated ELT antenna. The ELT aural tone should be heard on the AM broadcast receiver.

(b) This is not a measured check, but it does provide confidence that the antenna is radiating sufficient power to aid search and rescue.

NOTE: Use of the aircraft's VHF receiver, tuned to 121.5 MHz, is not recommended. This receiver is more sensitive and could pick up a weak signal even if the radiating ELT antenna is disconnected. While it will confirm the ELT is active, it does not check the integrity of the ELT system.

(3) G-switch Check

Test operation of the G-switch annually as follows:

CAUTION: EVEN WITHOUT AN ANTENNA CONNECTED, THE SIGNAL CAN BE RECEIVED BY A SATELLITE. AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

(a) Setup

1) The ELT must be removed from the airplane. See “Removal” on page 256019.

2) Fabricate a “shorting plug” by obtaining a standard 15-pin D-sub connector and installing a jumper (i.e. - short) between Pins 5 and 12. (Artex sells a test plug for this purpose under part number 150-1130.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the “front connector” on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in the “Installed Transmitter Test (Self Test)” on page 256023.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver on 121.5 MHz, if desired). Verify activation by listening for the aural sweep tone on the receiver. The LED on the ELT housing will also begin to flash, indicating activation. For proof, the AM radio will verify that the RF output is operational.

3) Allow ELT to transmit only long enough to verify operation.

CAUTION: ENSURE ELT IS RESET BEFORE 50 SECONDS HAS ELAPSED FROM SUCCESSFUL ACTIVATION. FAILURE TO DO SO WILL PERMIT A “LIVE” SIGNAL TO BE SENT TO THE SEARCH AND RESCUE SATELLITES.

4) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

CHART 3 (Sheet 1 of 2)
ELT 1000 SELF-TEST

Beeps/Flashes	Result	Indication/Action
1	System OK.	One pulse two seconds long of the LED, remote light and buzzer.
2	Code not used in ELT 1000.	
3	Code not used in ELT 1000.	
4		Low output power.
	Bad load or faulty ELT unit	1) Terminate antenna cable at antenna connection end w/50Ω load. Repeat self-test. If error clears proceed to step 3.
		2) Terminate antenna cable at antenna connection end w/50Ω load. Repeat self-test. If error clears proceed to step 3.
		3) Check coax cable and connectors for shorts and opens, bad center pin, etc. Disconnect, then reconnect cable. Repeat self-test.
		4) Check antenna ground for low resistance (should be <100 m Ω).
		5) Antenna should be 3 feet away from any vertical metal.
		6) Have antenna checked by factory.
5		No position data present (long message protocol only. Error is suppresed for short messages).
	Aircraft navigation system “OFF”.	Turn ON navigation system.
	Faulty system interface wiring or connections.	Check integrity of wiring and connections.
6		G-Switch loop between pins 5 and 12 is not present.
	Jumper open.	Verify D-Sub connector jumper is installed by checking for less than 1Ω between pins 5 and 12. Repair as necessary.
	Jumper missing.	Install G-SW enable jumper between pins 5 and 12 in D-sub harness connector.
7		Battery Check.
	Battery operating time is >1 hr.	Replace battery pack.
	Battery voltage low.	Replace battery pack.
	Battery memory read-write error.	Repeat self-test. Replace battery pack if 7-flash persists.

CHART 3 (Sheet 2 of 2)
ELT 1000 SELF-TEST

Beeps/Flashes	Result	Indication/Action
8		Protocol Programming Data.
	Missing data such as ELT S/N or aircraft tail number.	ELT requires reprogramming w/appropriate data.

(4) Digital Message Verification

The ARTEX ELT 1000 transmits a 406 MHz message upon reset, which is encoded such that it will be ignored by the SAR satellite system. The 15-digit number contained in this transmission is used to register the ELT with the appropriate 406 MHz ELT registration authority. In the US, the National Oceanic and Atmospheric Administration (NOAA) maintains the database of registered ELTs. The information in this database provides the Search and Rescue system with aircraft identification data in the event an actual distress signal is transmitted.

Verify the 406 MHz digital message as follows:

(a) Required Equipment

A test set capable of receiving and decoding the message - i.e., Artex ELT Test Set P/N 453-1000, or equivalent.

(b) Procedure

1) At right side of empennage, remove four (4) screws and remove ELT access panel.

2) Disconnect the antenna coax cable at the ELT, connect test set or terminate as applicable.

3) Perform all necessary steps to prepare Test Set to receive 406 MHz signal including (but not limited to) turning on power, activating program or any other steps required for the particular Test Set being used.

4) Perform the “Installed Transmitter Test (Self Test)” on page 256023.

5) Watch the screen on the Test Set to ensure that a message has been received. Repeat Self Test if necessary.

6) View message, ensure that all applicable information is correct (country code, aircraft ID, etc.).

7) The 15 digit ID hex ID (for example “ADC6492640D3411F1’) should match what is shown on the ELT product label. This is the 15 digit hex ID (Unique Identification Number or ‘UIN”) that is used to register the ELT.

8) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

H. Electrical Schematic

See 91-25-60.

I. Programming

(PIR-107994 Rev. C)

This procedure is used when replacing the Artex ELT1000 emergency locator transmitter.

NOTE: Emergency locator reprogramming is required on all ELT1000 equipped aircraft. Perform the following:

(1) Plug programming cable into the ELT and connect to the Handheld Programmer.

(2) Power on Handheld. Press “Home”. Main Screen View should appear.

NOTE: Press “Home” until Artex Icon appears on the Main Screen View.

(3) From the Handheld Programmer Main Screen View, Select “ARTEX”.

(4) Artex ELT Programmer page, select “Program”.

(5) Artex ELT Programmer page.

NOTE: Artex Programmer may default to previously entered Model, Protocol and Country Code.

(a) Dropdown menu, select ELT Model “ELT1000/345”

(b) Program GPS Baud 4800:

1) Select “Options”.

2) Select the “Test” position on the ELT 1000 and release within 2 seconds AFTER selecting “Get Options” on the Artex Programmer.

3) Select the Dropdown Menu.

4) Select the GPS Baud “4800”.

5) Select the “Test” Position on the ELT 1000 and release within 2 seconds AFTER selecting “Set” on the Artex Programmer.

6) Verify “Successfully Programmed Speed” is displayed on the Artex Programmer.

7) Select “Return”/.

(c) Dropdown menu Protocol

For U.S. aircraft and aircraft bound for export country that requires Tail number registration on the ELT:

- Select “Aviation User/Tail No./Loc (L)”.
- Dropdown menu, select desired Country Code.
- Enter the tail number of the aircraft (N number), which may by up to seven alphanumeric characters. Use stylus to select “abc” (for alphabet) and select “123” (for numbers). (If keyboard was opened, press “Done”)
- When finished, select “Enter”.

For aircraft bound for export country that requires 24 bit address registration:

- Select “Std Loc/Aircraft 24-bit Addr (L)”.
- Select the desired number system (Octal: 8 digits from 0-7 or Hex: 6 digits from 0-9 or A-F) to match the format of the 24 bit address provided.
- Enter the 24-bit address. (If keyboard was opened, press “Done”)
- When finished, select “Enter”.

NOTE: The 24 digit hex code is shown at the bottom of the screen with the abbreviated 15 digit hex code below it.

(d) Select the test position on the ELT 1000 switch and release. Then within 2 seconds, press the “Program” icon on the programmer.

(e) Verify that the screen shows “Programming Successful Verify ELT Using Sarcalc”.

(6) Label Printing

(a) Connect the Handheld Programmer to the applicable printer using the specified cable and adapter.

(b) After the ELT has been successfully programmed, select “Print”, from Printing ELT Data, select “Combined Label”. After printing is complete, select “Return”.

NOTE: Place new label over current ELT’s label that reads, “This ELT is Programmed For and Must be Registered in Country, Three Digit Country Code, 15-Digit Hex ID”.

(7) Programming Verification

NOTE: For a Direct Verification, connect the Handheld coax cable to the ELT and for a Remote Verification, remove coax cable and connect Handheld Whip Antenna. For a Remote Verification, a distance of 0-30 feet from the ELT antenna is recommended.

(a) Select “Sarcalc”.

(b) Select “Measure”.

(c) The screen will display “Waiting for Data”.

(d) Turn the ELT switch to “TEST” then release.

(e) Verify the message contains the correct information for the aircraft.

(f) When completed, turn the Handheld OFF.

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CHAPTER

26

FIRE PROTECTION

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CHAPTER 26

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
26-List of Effective Pages	1	Nov 30/17			
	2	Sep 15/09			
26-Table of Contents	1	Nov 30/17			
	2	Sep 15/09			
26-20-00	1	Nov 30/17			
	2	Nov 30/17			
	3	Nov 30/17			
	4	Nov 30/17			
	5	Nov 30/17			
	6	Nov 30/17			

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CHAPTER 26 - FIRE PROTECTION

TABLE OF CONTENTS

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EXTINGUISHING

WARNING: AFTER DISCHARGE OF EXTINGUISHER AVOID EXPOSURE TO SMOKE, VAPORS AND OTHER BY-PRODUCTS OF FIRE. DO NOT INCINERATE EXTINGUISHER.

CAUTION: EXTINGUISHER IS A PRESSURE VESSEL. PROTECT FROM CORROSIVE CONDITIONS. IF THERE IS ANY CORROSION OR DAMAGE, EXTINGUISHER SHOULD BE CAREFULLY EMPTIED AND DISCARDED. USE ONLY AS DIRECTED.

Portable Fire Extinguisher

A. Disposable, Class 2B:C Fire Extinguisher Model No. RT-A600

(1) Description

Containing 1.2 lbs. (550 grams) of Halon 1211/1301 blend, the extinguisher is located either in the top of the right-hand pyramid cabinet (if installed) or on the floor just aft of the spar carry-thru and behind the co-pilot's seat (see "Figure 3" on page 26204). When floor-mounted, the extinguisher is secured by a quick-release bracket.

To operate the extinguisher, remove it from the quick-release bracket, hold it upright in either hand by the handgrip, with the spray nozzle pointing forward. Remove the safety pin, direct the nozzle towards the base of the fire source, depress the lever. Maximum extinguishing effect is obtained if the fire fighter uses side to side motion and keeps moving in towards the base of the fire source as it is extinguished. Releasing the lever closes a secondary seal inside the operating head. This interrupts the flow of extinguishant, thus retaining part of the charge, for dealing with a flash back or re-ignition should they occur, without waste or leakage. A partly or totally discharged extinguisher should be replaced immediately after use.

(2) Inspection

Disposable type fire extinguishers should be maintained and inspected in accordance with the nameplate instructions.

(a) Before Each Flight

RT-A600 extinguishers with date codes 2012 and later are subject to H3R Service Bulletin No. 2015-01 (Piper Service Bulletin No. 1280A). These extinguishers should be labeled "Weigh this unit before each flight" and must be weighed before each flight. Weight must be above the minimum specified on the nameplate.

(b) Monthly

Inspect monthly or more frequently. Ensure nozzle is not obstructed and safety seal is intact. Inspection is a "quick check" that an extinguisher is available and will operate. It is intended to give reasonable assurance that the extinguisher is fully charged and operable. This is done by seeing that it is in its designated place, that it has not been actuated (discharged) or tampered with, and that there is no obvious physical damage or condition to prevent operation. Determine fullness by weighing or "hefting."

Figure 1
Model RT-A600
(Piper P/N 459-887)

(c) 100 Hour / Annual

Each 100 hours or annually, whichever comes first, weigh the extinguisher. Replace and return to manufacturer if gross weight is below the minimum specified on the nameplate.

B. Gauged, Class 2B:C Fire Extinguisher Model No. A344T

(1) Description

The extinguisher contains 1.25 lbs. (567 grams) of Halon 1211. When factory installed, the extinguisher is located on the floor just aft of the spar carry-thru and behind the co-pilot's seat (see "Figure 3" on page 26204). It is secured by a quick release bracket.

NOTE: In airplanes equipped with a right side pyramid cabinet, if the original RT-A600 extinguisher has been field replaced with an A344T extinguisher, the mounting location will vary. The RT-A600 was installed in the pyramid cabinet. The A344T is too large to fit in the same location, so, for those aircraft, some other FAA approved method of mounting the fire extinguisher would have been used when complying with Piper Service Bulletin No. 1280A, or later revision.

To operate the extinguisher, remove it from the quick-release bracket, hold it upright in either hand by the handgrip, with the spray nozzle pointing forward. Remove the safety pin, direct the nozzle towards the base of the fire source, depress the lever. Maximum extinguishing effect is obtained if the fire fighter uses side to side motion and keeps moving in towards the base of the fire source as it is extinguished. Releasing the lever closes a secondary seal inside the operating head. This interrupts the flow of extinguishant, thus retaining part of the charge, for dealing with a flash back or re-ignition should they occur, without waste or leakage. A partly or totally discharged extinguisher should be replaced immediately after use.

Figure 2
Model A344T
(Piper P/N 459-750)

Gauged type fire extinguishers should be installed, inspected, and maintained in accordance with the owner's manual supplied with the extinguisher and NFPA No. 10, "Portable Fire Extinguishers." The following summarizes portions of those two documents.

(2) Inspection

Check when initially placed in service and at minimum 30-day intervals or more frequently if circumstances dictate. Persons performing 30-day inspections are not be required to be certified.

Check to see that the extinguisher is undamaged; that the nozzle is unobstructed; that the gauge pressure is in the operable (green) range; that the lockpin and tamper seal is in place, and that the operating instructions are clearly visible. If the inspection reveals a deficiency in any of these conditions, corrective action must be taken.

(3) Maintenance

(a) Annual

Extinguishers shall be subjected to maintenance not more than one year apart or when specifically indicated by an inspection. Maintenance is a thorough examination of the extinguisher, covering mechanical parts, extinguishing agent and expelling means. It is intended to give maximum assurance that an extinguisher will operate effectively and safely, and should be done professionally. Most authorities require special tags be attached to the extinguisher to verify this service.

(b) Six-Year Maintenance

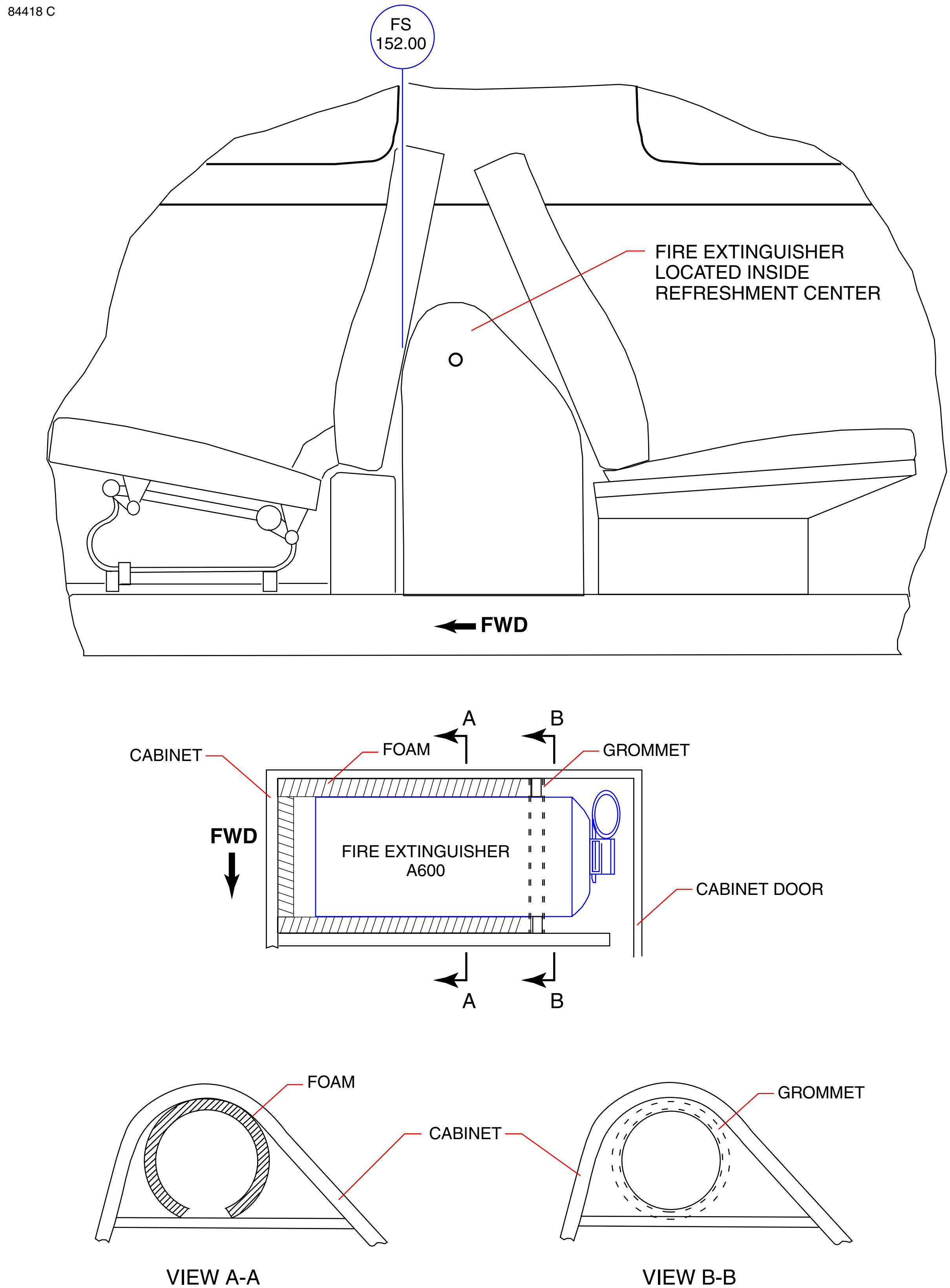
Every six years, stored pressure fire extinguishers that require a 12-year hydrostatic test shall be emptied and subjected to applicable maintenance procedures. This should be done professionally, and involves a thorough inspection and the replacement of certain parts. When these maintenance procedures are performed during periodic recharging or hydrostatic testing, the six year requirement shall begin from that date.

(c) Twelve-Year Hydrostatic Test

All gauged H3R Aviation fire extinguishers require a 12 year hydrostatic test performed by certified technicians as per NFPA 10. Simply stated, the Hydrostatic test confirms the integrity of the cylinder, and at the 12 year mark, is performed in conjunction with the 6 year maintenance.

C. Replacement Intervals

Replace fire extinguishers as specified in 5-30-00.



Fire Extinguisher Locator
Figure 3 (Sheet 1 of 2)

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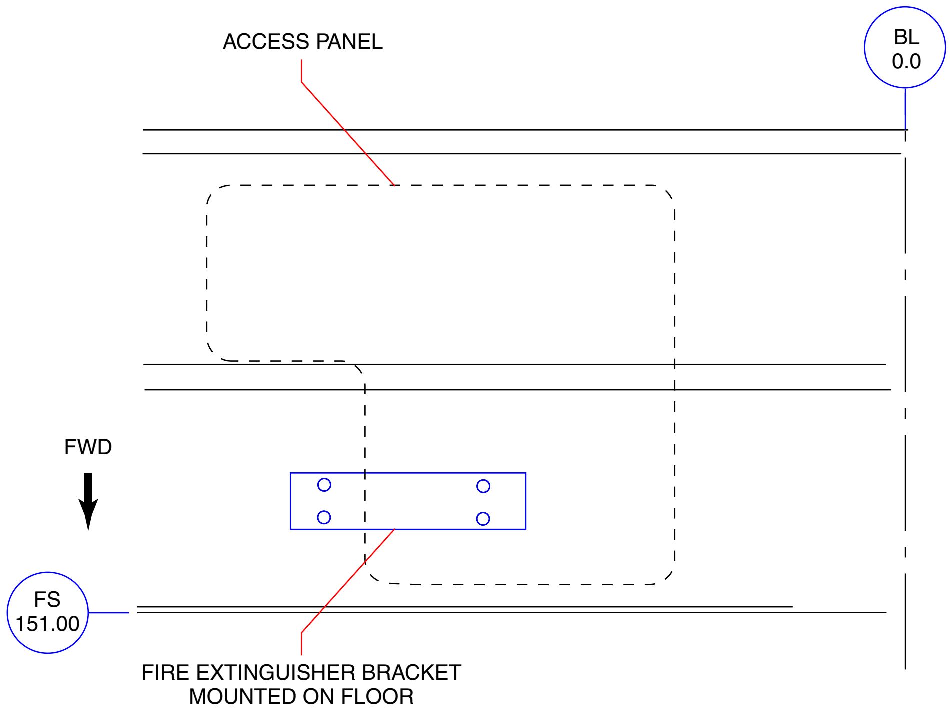
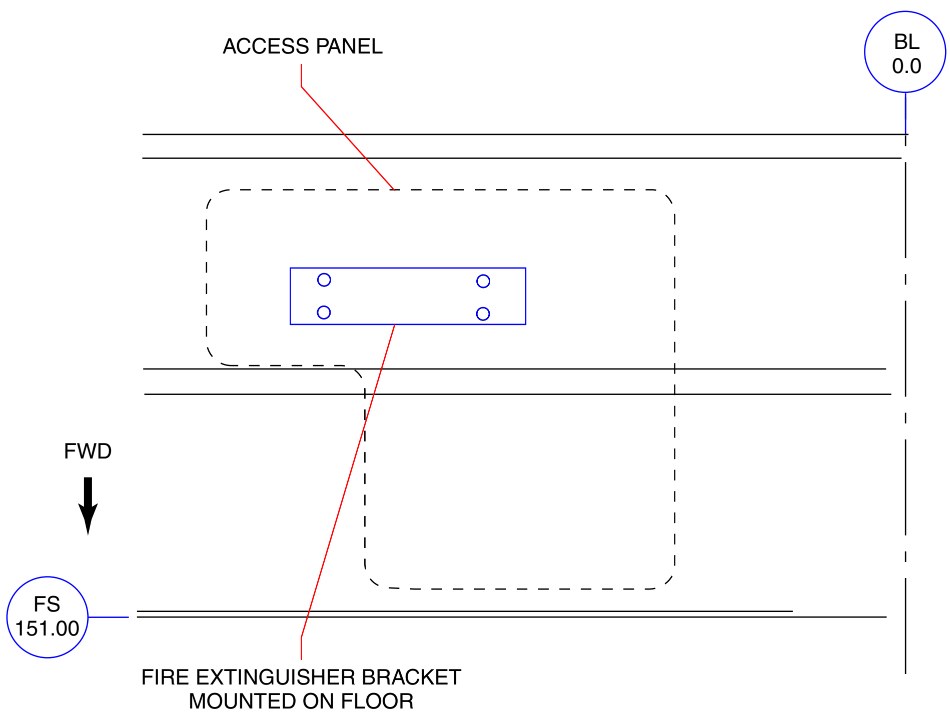


PA-46-350P



PA-46R-350T

Fire Extinguisher Locator
Figure 3 (Sheet 2 of 2)

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CHAPTER

27

FLIGHT CONTROLS

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CHAPTER 27

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
27-List of Effective Pages	1	Mar 25/21
	2	Mar 25/21
27-Table of Contents	1	Mar 25/21
	2	Mar 25/21
	3	Mar 25/21
	4	Sep 15/09
27-00-00	1	Sep 15/09
	2	Sep 15/09
	3	Dec 15/14
	4	Feb 15/14
	5	Nov 30/17
	6	Sep 15/09
	7	Sep 15/09
	8	Sep 15/09
	9	Jan 15/13
	10	Sep 15/09
	11	Sep 15/09
	12	Sep 15/09
27-10-00	1	Nov 30/17
	2	Sep 15/09
	3	Sep 15/09
	4	Nov 30/17
	5	Sep 15/09
	6	Jul 31/19
	7	Jul 31/19
	8	Nov 30/17
	9	Nov 30/17
	10	Nov 30/17
	11	Nov 30/17
	12	Nov 30/17
	13	Nov 30/17
	14	Nov 30/17
	15	Jul 31/19
	16	Nov 30/17
	17	Mar 25/21
	18	Mar 25/21
	19	Mar 25/21
	20	Mar 25/21
	21	Mar 25/21
	22	Mar 25/21
27-20-00	1	Sep 15/09
	2	Nov 30/17
	3	Sep 15/09
	4	Nov 30/17
	5	Mar 25/21
	6	Mar 25/21
	7	Feb 15/14
	8	Mar 25/21
	9	Mar 25/21
	10	Mar 25/21
27-30-00	1	Sep 15/09
	2	Sep 15/09
	3	Nov 30/17
	4	Nov 30/17
	5	Nov 30/17
	6	Mar 25/21
	7	Nov 30/17
	8	Nov 30/17
	9	Nov 30/17
	10	Nov 30/17
	11	Jul 31/19
	12	Mar 25/21
	13	Mar 25/21
	14	Mar 25/21
	15	Mar 25/21
	16	Mar 25/21
	17	Mar 25/21
	18	Mar 25/21
	19	Mar 25/21
	20	Mar 25/21
	21	Mar 25/21
	22	Mar 25/21
	23	Mar 25/21
	24	Mar 25/21

CHAPTER 27

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
27-50-00	1	Nov 30/17			
	2	Dec 15/14			
	3	Nov 30/17			
	4	Nov 30/17			
	5	Dec 15/14			
	6	Sep 15/09			
	7	Dec 15/14			
	8	Nov 30/17			
	9	Nov 30/17			
	10	Nov 30/17			
	11	Mar 25/21			
	12	Mar 25/21			
	13	Mar 25/21			
	14	Dec 15/14			
27-60-00	1	Sep 15/09			
	2	Mar 25/21			
	3	Sep 15/09			
	4	Nov 30/17			
	5	Sep 15/09			
	6	Sep 15/09			

CHAPTER 27 - FLIGHT CONTROLS

TABLE OF CONTENTS

CHAPTER 27 - FLIGHT CONTROLS

TABLE OF CONTENTS (continued)

CHAPTER 27 - FLIGHT CONTROLS

TABLE OF CONTENTS (continued)

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GENERAL

1. Description and Operation

The airplane is controlled in flight by the use of three primary control surfaces, consisting of ailerons, elevator and rudder. Operation of these controls is through movement of the dual control wheels and rudder pedals. On the forward end of each control wheel is a quadrant assembly to which the aileron control cables are fastened. Also connected to the control wheel push-pull tubes are interconnect arms to the elevator torque tube located behind the instrument panel. Mounted on this torque tube are sectors to which the elevator control cables are fastened; thus the forward and aft movement of the control wheels in turn moves the torque tube and sectors thus transmitting movement thru the control cables to the elevator.

The rudder controls are driven from dual rudder pedals through torque tubes which have dual sectors connected and control cables from these sectors to the rudder sector located in the aircraft tail section. Provisions for elevator and rudder trim are provided by adjustable trim mechanisms. The trim controls and indicators are in the pedestal located on the instrument panel. Elevator trim is through cables routed to the elevator trim screw (located in the tail cone), which drives dual actuators to the trim tab. The rudder trim is controlled by a trim wheel and screw assembly connected to a bungee assembly connected to the rudder torque tube behind the instrument panel.

The ailerons and rudder are interconnected through a spring system, which is activated only when controls are out of harmony. In normal coordinated flight it is inactive.

2. Standard Practices and Procedures

A. General

NOTE: For additional information, refer to Federal Aviation Administration Advisory Circular AC 43.13-1B, Chapter 7. Download from the FAA at http://www.airweb.faa.gov/.

WARNING: FAILURE TO VERIFY FREE AND CORRECT OPERATION OF THE FLIGHT CONTROL SYSTEM MAY RESULT IN THE DESTRUCTION OF THE AIRPLANE AND LOSS OF LIFE, IF THE AIRPLANE IS FLOWN.

IF THERE IS ANY UNCERTAINTY, DO NOT FLY AN AIRPLANE UNTIL FREE AND CORRECT OPERATION OF THE FLIGHT CONTROL SYSTEM IS VERIFIED.

CONTACT PIPER CUSTOMER SERVICE FOR ASSISTANCE AT (+1) 772-299-2141 OR CUSTOMERSERVICE@PIPER.COM. PIPER'S NORMAL BUSINESS HOURS ARE MONDAY THROUGH FRIDAY, 7:30 A.M. TO 4:30 P.M. (EASTERN).

(1) It is important to remove all necessary access panels to gain both physical and visual access to control cables to ensure and verify proper routing in accordance with applicable service or maintenance manual instructions.

(2) Pay particular attention to control cable part numbers per the applicable parts catalog, in order to verify the proper replacement. Route and locate cables per the applicable service or maintenance manual.

(3) Where practical, remove and install one flight control cable, motor, actuator, or pushrod at a time. This will help avoid incorrect reassembly and aid in the proper connection of the flight controls.

(4) Flight control cable rigging tensions specified in "Chart 2" on page 27004; must be corrected to account for ambient temperature, use "Chart 1" on page 27003.

(5) Verify that the flight control system operates smoothly throughout the full range of motion.

(6) Visually confirm that all pulleys rotate freely. Visually confirm that all cables pass through all guides and fairleads without snagging or binding, and do not interfere with or chafe on any airframe or system components.

(7) Verify that the flight control surfaces move in the proper direction when commanded.

(8) Recheck the tension of replacement flight control cables upon reaching 50 hours of operation.

B. Tips for Removal, Installation, and Rigging of Various Assemblies

(1) It is recommended, though not always necessary, to level and place the airplane on jacks during rigging and adjustment.

(2) Remove turnbuckle barrels from cable ends before withdrawing the cables through the structures.

(3) Tie a cord to the cable end before withdrawing the cable through the structures to facilitate installation of cable.

(4) When referring to marking cable end, etc., before disconnecting, use a felt tip marking pen.

(5) Assemble and adjust the turnbuckles so that each terminal is screwed an approximately equal distance into the barrel. Do not turn the terminals in such a manner that will put a permanent “twist” into the cables.

(6) Cable tensions should be taken with the appropriate control surface in its neutral position.

(7) After completion of each adjustment, check the turnbuckles to be sure not more than three terminal threads are visible outside the barrel. Install the locking clips, and check for proper installation by trying to remove the clips using fingers only. Both locking clips may be installed in opposite holes. Locking clips which have been installed and removed must be scrapped and not reused. Turnbuckles may be safetied in accordance with Advisory Circular 43.13-1, latest revision.

(8) When pushrods or rod ends are provided with an inspection hole, the screw must be screwed in far enough to pass the inspection hole. This can be determined visually or by feel, inserting a piece of wire into the inspection hole. If no hole is provided, there must be a minimum of .375 of an inch thread engagement.

(9) When installing rod end jamnuts, refer to 20-00-00, Figure 2, for proper installation method.

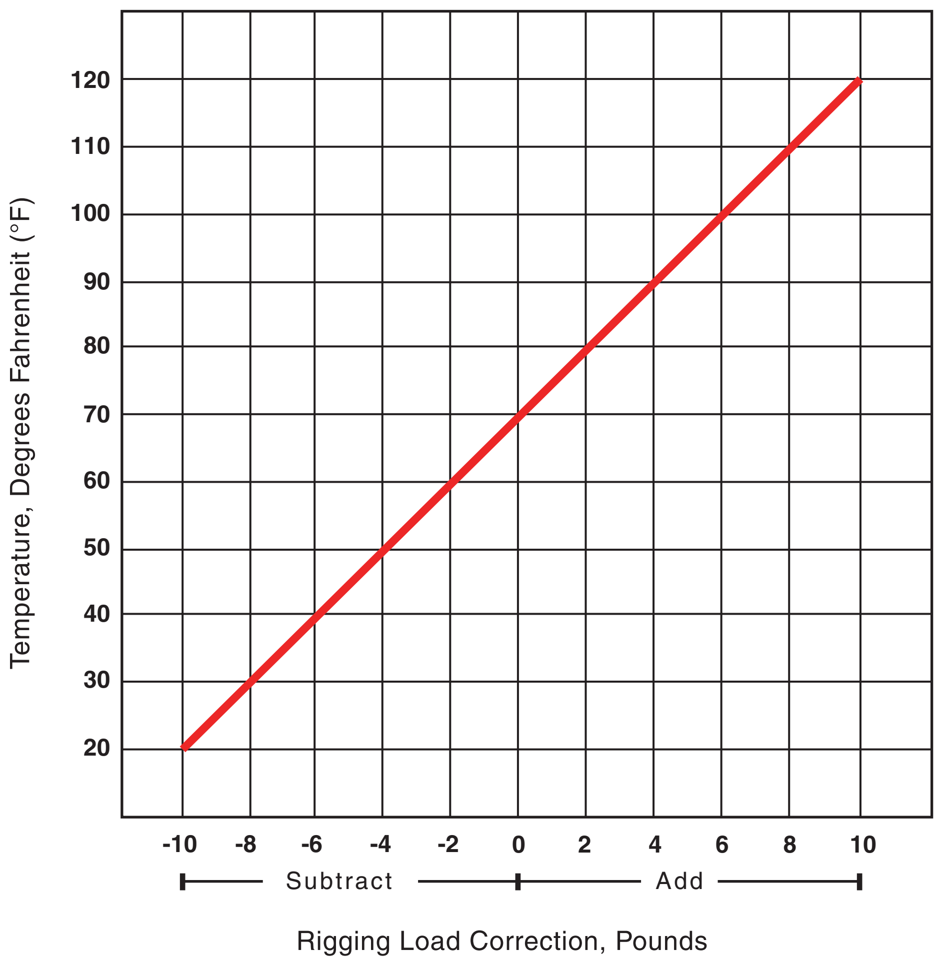
(10) After completion of adjustments, each jam nut must be tightened securely.

NOTE: Cable rigging tensions specified must be corrected to ambient temperature in the area where the tension is being checked, using “Chart 1”. On new airplanes, it is suggested that cables initially be tensioned 25 to 30 percent over nominal tension, then loosened to the “High Side” of the tolerance. This will aid in maintaining specified tension after flight testing.

(11) Torque all nuts in the flight control system (including nose wheel steering), refer to Recommended Nut Torques Chart in Chapter 91.

(12) Check all cable ball ends for proper seating in retainers after setting cable tension.

CHART 1
CABLE TENSION VS. AMBIENT TEMPERATURE



3. Flight Control Surface Travel

   See "Chart 2" for specifications, see appropriate section for rigging instructions.

4. Flight Control Cable Tension

   CAUTION: CABLE TENSIONS GIVEN IN "CHART 2" APPLY ONLY TO AIRPLANES WITHOUT AUTOPILOT BRIDLE CABLE INSTALLATIONS, UNLESS THE AUTOPILOT SYSTEM WAS FACTORY INSTALLED BY PIPER. IF AN AUTOPILOT USING BRIDLE CABLES HAS BEEN INSTALLED AFTER LEAVING THE FACTORY, CONSULT THE APPROPRIATE AUTOPILOT VENDOR DOCUMENTATION FOR CORRECT CABLE TENSIONS WITH AUTOPILOT BRIDLE CABLES ATTACHED.

   A. See "Chart 2" for specifications, see appropriate section for rigging instructions.

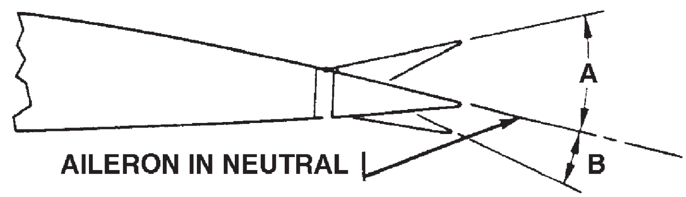
   B. When a new cable is installed, cable tension must be rechecked after flight test.

CHART 2 (Sheet 1 of 2)
FLIGHT CONTROL SURFACES RIGGING LIMITS

PPS-50056-2, Rev. AA

AILERONS

A = 18° ± 1° UP
B = 18° ± 1° DOWN



CABLE TENSION = 30 LBS ± 5 LBS

NOTE
1. MAXIMUM FREE PLAY SHALL NOT EXCEED 0.120 INCHES MEASURED AT THE AILERON INBOARD TRAILING EDGE.

FLAPS

A = 0° ± 1° (NEUTRAL)
B = 10° + 2°, - 2° (FIRST NOTCH OF TRAVEL)
C = 20° + 2°, - 0° (SECOND NOTCH OF TRAVEL)
D = 36° + 0°, - 1° (THIRD NOTCH OF TRAVEL)

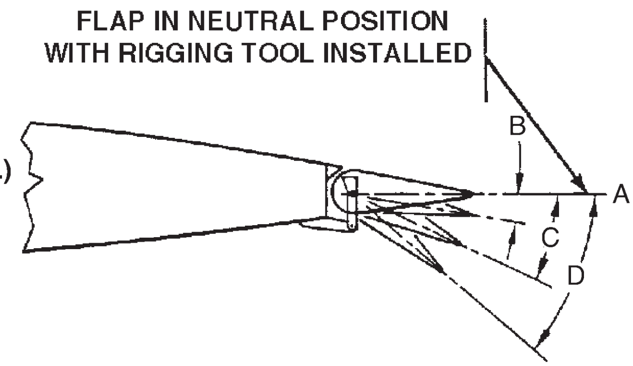


CHART 2 (Sheet 2 of 2)
FLIGHT CONTROL SURFACES RIGGING LIMITS

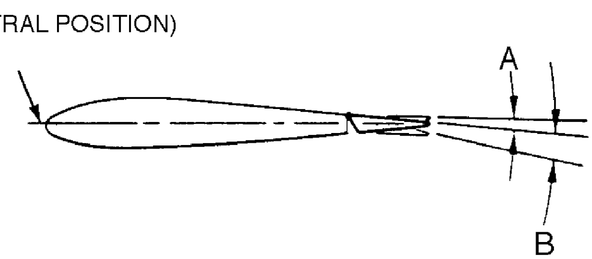
ELEVATOR

A = 23.5° ± .5° TRAILING EDGE UP
B = 14.5° ± .5° TRAILING EDGE DOWN

CABLE TENSION = 30 LBS ± 5 LBS

ELEVATOR TRIM TAB

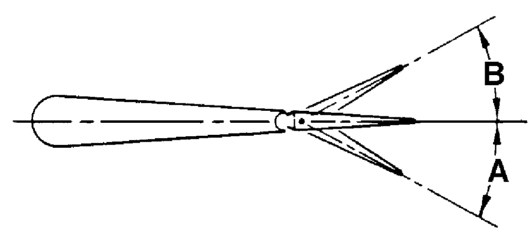
A = 19.0° + 0°, - 1° TAB TRAILING EDGE UP
B = 24.5° + 0°, - 1° TAB TRAILING EDGE DOWN



CABLE TENSION = 12 LBS ± 2 LBS

RUDDER

A = 26° + 1°, - 0° LEFT
B = 30° + 1°, - 0° RIGHT



CABLE TENSION = 40 LBS ± 5 LBS

LOOKING DOWN

(SEE 32-20-00, FIGURE 5, FOR RUDDER PEDAL NEUTRAL ANGLE)

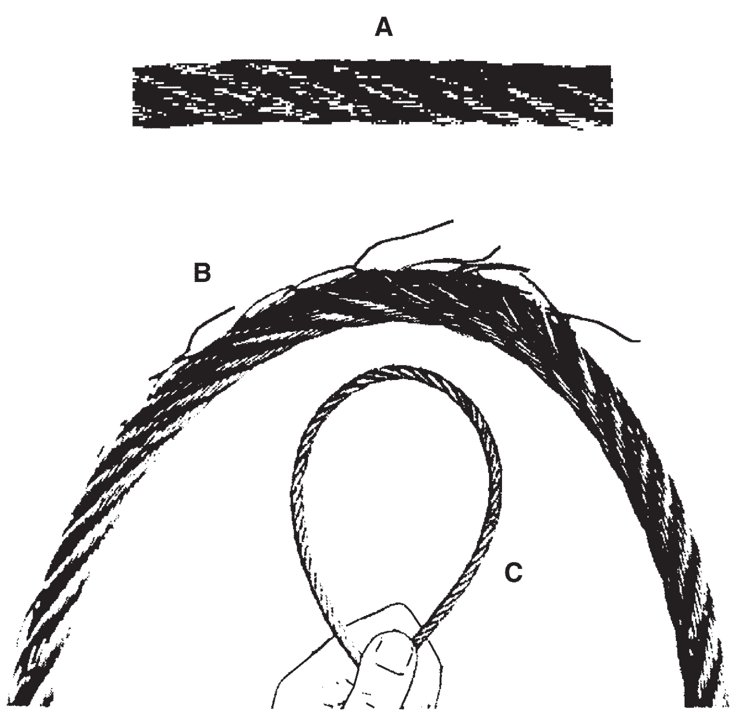
5. Control Cable Inspection

Aircraft control cable systems are subject to a variety of environmental conditions and forms of deterioration that, with time, may be easy to recognize as wire/strand breakage or the not-so-readily visible types of wear, corrosion, and/or distortion. The following data may help in detecting the presence of these conditions:

A. Cable Damage

Critical areas for wire breakage are sections of the cable which pass through fairleads and around pulleys. To inspect each section which passes over a pulley or through a fairlead, remove cable from aircraft to the extent necessary to expose that particular section. Examine cables for broken wires by passing a cloth along length of cable. This will clean the cable for a visual inspection, and detect broken wires, if the cloth snags on cable. When snags are found, closely examine cable to determine full extent of damage.

The absence of snags is not positive evidence that broken wires do not exist. “Figure 1”A shows a cable with broken wires that were not detected by wiping, but were found during a visual inspection. The damage became readily apparent (“Figure 1”B) when the cable was removed and bent using the techniques depicted in “Figure 1”C.



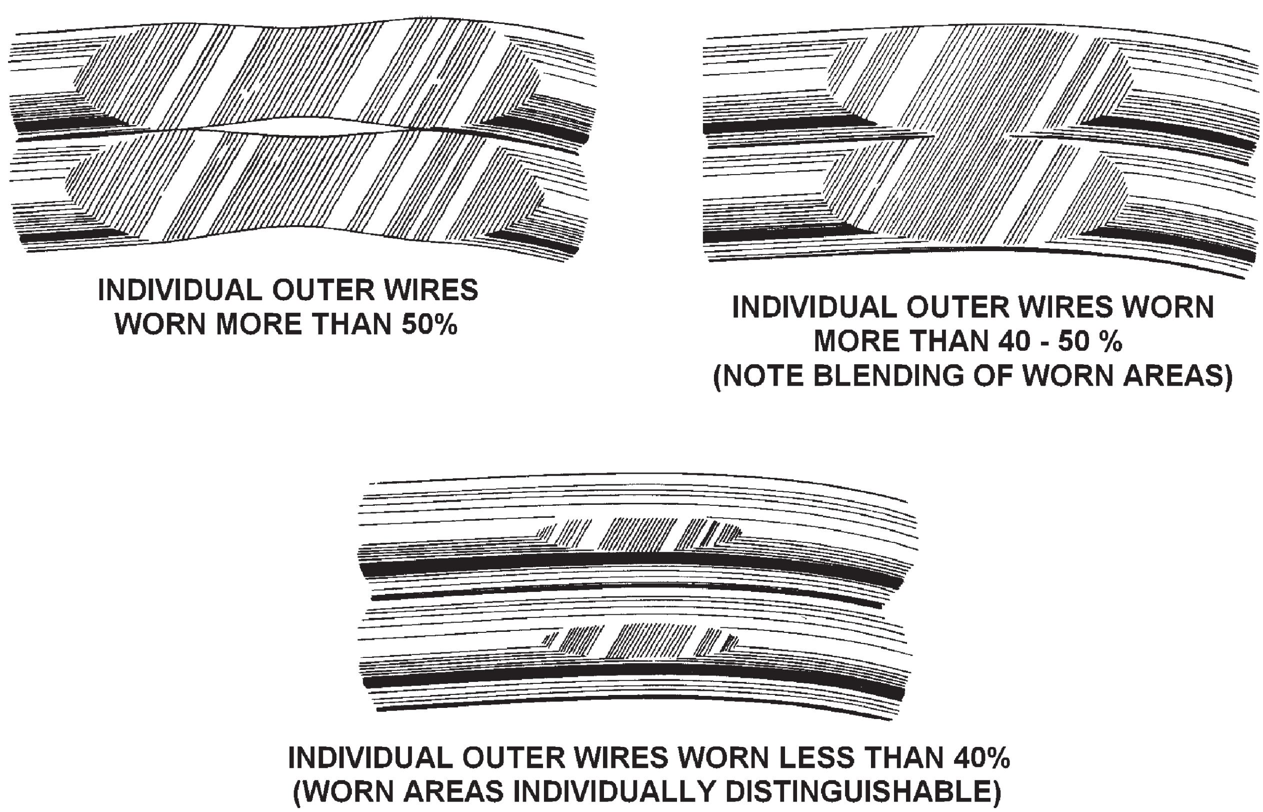
Control Cable Inspection
Figure 1

B. External Wear Patterns

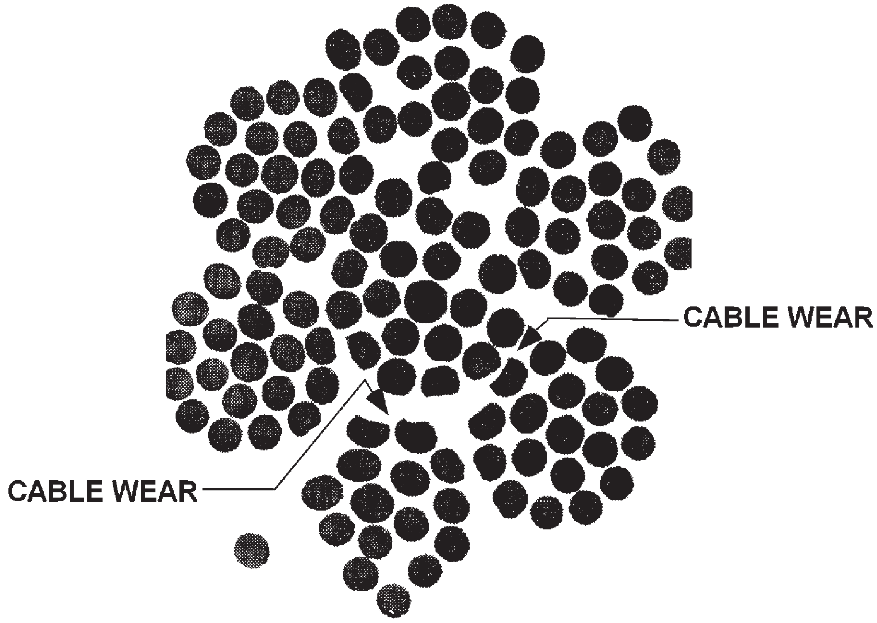
Wear will normally extend along cable equal to the distance cable moves at that location. Wear may occur on one side of the cable only or on its entire circumference. Replace flexible and non-flexible cables when individual wires in each strand appear to blend together (outer wires worn 40-50 percent) as depicted in “Figure 2”.

C. Internal Cable Wear

As wear is taking place on the exterior surface of a cable, the same condition is taking place internally, particularly in the sections of the cable which pass over pulleys and quadrants. This condition, shown in “Figure 3”, is not easily detected unless the strands of the cable are separated. Wear of this type is a result of the relative motion between inner wire surfaces. Under certain conditions the rate of this type wear can be greater than that occurring on the surface.



External Cable Patterns
Figure 2



Internal Cable Patterns
Figure 3

D. Corrosion

Carefully examine any cable for corrosion that has a broken wire in a section not in contact with wear-producing airframe components such as pulleys, fairleads, etc. It may be necessary to remove and bend the cable to properly inspect it for internal strand corrosion, as this condition is usually not evident on the outer surface of the cable. Replace cable segments if internal strand rust or corrosion is found.

Areas especially conducive to cable corrosion are battery compartments, lavatories, wheel wells, etc., where concentrations of corrosive fumes, vapors, and liquids can accumulate.

NOTE: Check all exposed sections of cable for corrosion after a cleaning and/or metal-brightening operation has been accomplished in that area.

E. Cable Maintenance

CAUTION: TO AVOID REMOVAL OF CORROSION-PREVENTATIVE COMPOUNDS AND CABLE INTERNAL LUBRICANT, DO NOT USE VAPOR DEGREASING, STEAM CLEANING, METHYLETHYLKETONE (MEK) OR OTHER SOLVENTS.

CAUTION: DO NOT OIL CONTROL CABLES.

Frequent inspections and preservation measures such as rust prevention treatments for bare cable areas will help to extend cable service life. Where cables pass through fairleads, pressure seals, or over pulleys, remove accumulated heavy coatings of corrosion prevention compound. Provide corrosion protection for these cable sections by lubricating as specified in the Lubrication Chart, 12-20-00.

F. Cable Fittings (Terminals, turnbuckles, etc.)

Clean the fittings thoroughly with a suitable solvent prior to inspection.

NOTE: Most control cables feature strands made of galvanized steel. By design, the galvanized coating corrodes sacrificially, protecting the steel strands underneath. This normal corrosion process creates zinc oxide powder, which can migrate along the length of the control cable, covering the surface of the fittings with an inert white speckled coating. This coating, along with any other contaminants such as dirt, oil or grease, must be removed in order to properly examine the fittings.

(13) 100 Hour Standard Inspection

Check swaged terminal reference marks for any indication of cable slippage within fitting. Inspect fitting assembly for distortion and/or broken strands at the terminal. Check that all bearings and swivel fittings (bolted or pinned) pivot freely to prevent binding and subsequent failure. Check turnbuckles for proper thread exposure and broken or missing safety wires/clips.

Pay particular attention to corrosion and “pitting” on cable terminals, turnbuckles and cable fittings. Any corrosion or pitting found requires replacement of the corroded fitting and/or cable.

(14) 100 Hour Special Inspection

For airplanes 15 years old or older, using a 10X magnifier, visually inspect the entire surface of each cable terminal, turnbuckle, or other cable fitting for corrosion or cracking. Inspect under safety wire or clips wrapped around the cable or fitting. Any evidence of corrosion or cracking, however minute, is cause for replacement. A logbook entry documenting the replacement of a cable terminal, turnbuckle, or other cable fitting relieves the inspection requirement for that fitting only, until such time as that fitting has been in service for 15 years.

G. Pulleys

Inspect pulleys for roughness, sharp edges, and presence of foreign material embedded in the grooves. Examine pulley bearings to assure proper lubrication, smooth rotation, freedom from flat spots, dirt, and paint spray. Periodically rotate pulleys, which turn through a small arc, to provide a new bearing surface for the cable. Maintain pulley alignment to prevent the cable from riding on flanges and chafing against guards, covers, or adjacent structure. Check all pulley brackets and guards for damage, alignment, and security.

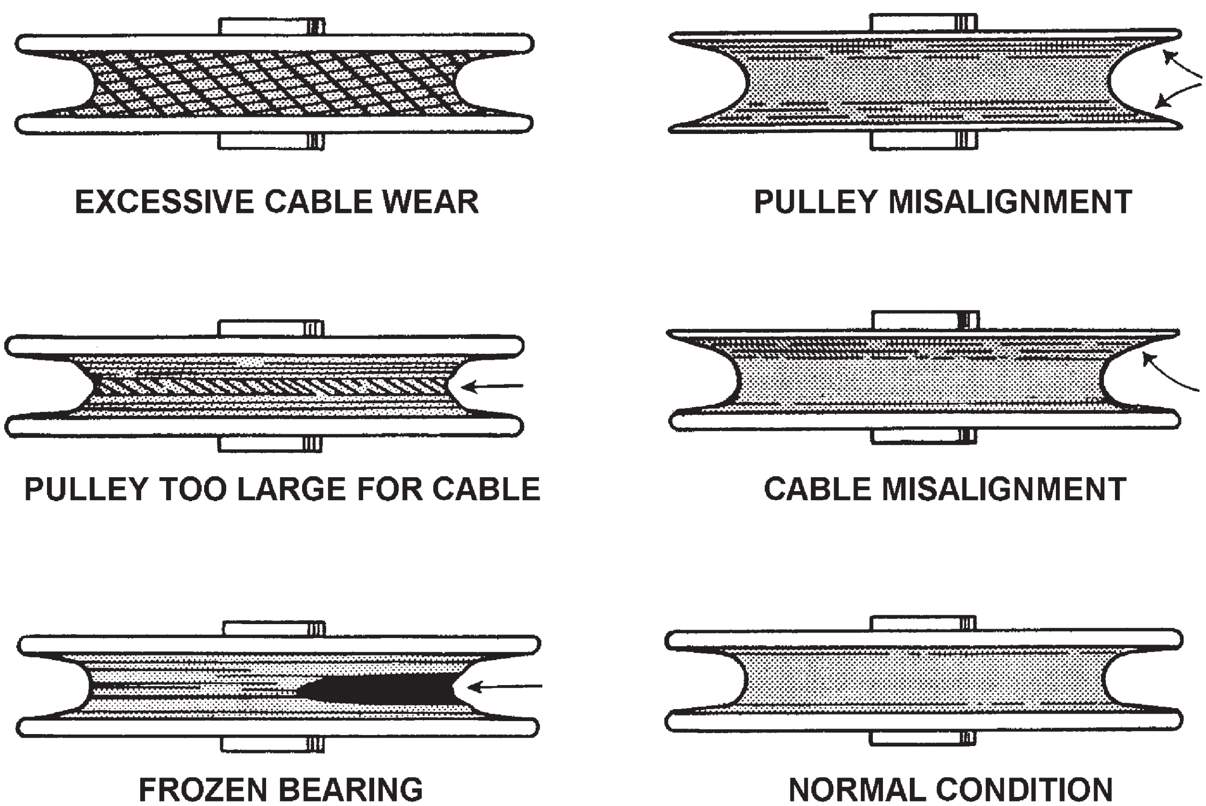
H. Pulley Wear Patterns

Various cable system malfunctions may be detected by analyzing pulley conditions. These include such discrepancies as too much tension, misalignment, pulley bearing problems, and size mismatches between cables and pulleys. Examples of these conditions are shown in “Figure 4”.

6. Special Tools

These tools are used at the factory and are recommended where the following rigging and/or alignment procedures specify “use a suitable tool.” Order them through any Piper Dealer.

Tool (Tool Number)	Piper P/N
Rudder Pedal Neutral Alignment Tool (HF 622724)	762-151
Elevator / Elevator Trim Travel Tool (HF 689259)	762-144
Nose Wheel Centering (i.e. - Neutral) Tool (HF 622722)	762-147
Aileron and Flap Travel Tool - Left Wing (CKJ 622719)	762-149
Aileron and Flap Travel Tool - Right Wing (CKJ 62)	762-150
Rudder Travel Tool (TL 622721)	762-153
Control Wheel Neutral Tool (HF 622725)	N/A



Pulley Wear Patterns
Figure 4

7. Fin-Type Control Cable Air Seals

(Refer to "Figure 5".)

This airplane is equipped with ten fin type control cable air seals. Six seals (one for each control cable going to the empennage) are located on the aft pressure bulkhead. Two seals are located in each wing root for the aileron cables.

A. Drawing Cables Through

It is not necessary to completely remove the seal from the bulkhead in order to draw a cable turnbuckle through the seal.

To draw a cable turnbuckle through the seal.

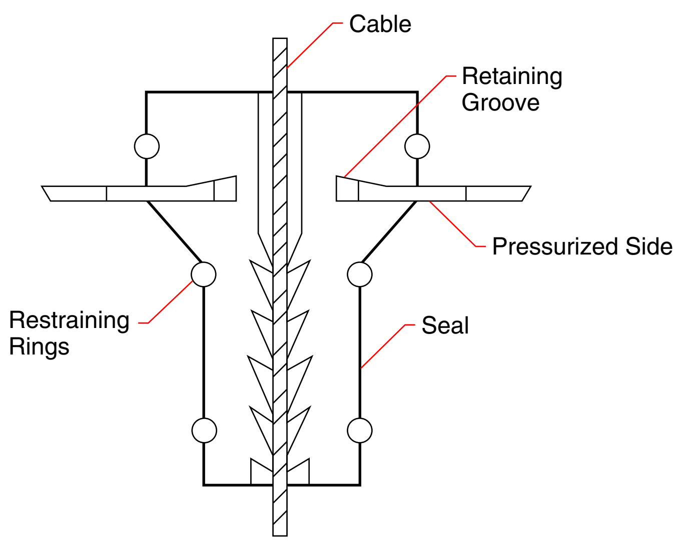
(1) Remove the three seal clips (per seal).
(2) Grease the cable and turnbuckle with MIL-G-21164 grease.
(3) Slowly pull the cable through the seal.

B. Removal

(1) Remove the three seal clips (per seal).
(2) Compress the seal and pull it from the hole in the bulkhead.
(3) Remove the seal from the cable.

C. Installation

(1) Fill the cable air seal with MIL-G-21164 grease.
(2) Apply MIL-G-21164 grease to the full travel of the cable section that passes through the seal when the control is actuated.
(3) Apply a light coat of MIL-G-21164 on the external small end and tapered section of the seal for ease of insertion.
(4) Place the seal on the cable, on the non-pressurized side of the bulkhead with the small end of the seal pointing towards the bulkhead.
(5) Insert the seal in the bulkhead hole so that the bulkhead sheet is entirely within the seal retaining groove.
(6) Install the proper retaining rings on the seal, two on the small end and one on the large end.



Fin Type Control Cable Air Seal
Figure 5

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AILERON AND TAB

1. Troubleshooting

   See"Chart 1" on page 27102.

2. Control Wheel

   A. S/N's 4636001 thru 4636020 only.

      NOTE: In these airplanes, the control wheel and tube can be removed as a unit, see Control Column, S/N's 4636001 thru 4636020, below. Disassembly of the control wheel from the tube is not normally required.

      (1) Removal

         (a) Remove two screws from bottom of control wheel and remove cover/pad assembly to gain access to locknut.

         (b) Remove control wheel attachment bolt, spacer, washer, and locknut. Note build up to facilitate reinstallation.

         (c) Slide control wheel off of tube.

         (d) If not already done, disconnect control wheel switches wiring harness.

      (2) Installation

         (a) Connect control wheel switches wiring harness.

         (b) Slide control wheel onto tube.

         (c) Position and secure control wheel attachment bolt, spacer, washer, and locknut.

         (d) Position cover/pad assembly and secure with two screws from bottom of control wheel.

   B. S/N's 4636021 thru 4636131 only.

      (1) Removal

         (a) Remove two screws from bottom of control wheel and remove cover/pad assembly to gain access to locknut.

         (b) Remove control wheel attachment bolt, spacer, washer, and locknut. Note build up to facilitate reinstallation.

         (c) Slide control wheel off of tube.

         (d) If not already done, disconnect control wheel switches wiring harness.

      (2) Installation

         (a) Connect control wheel switches wiring harness.

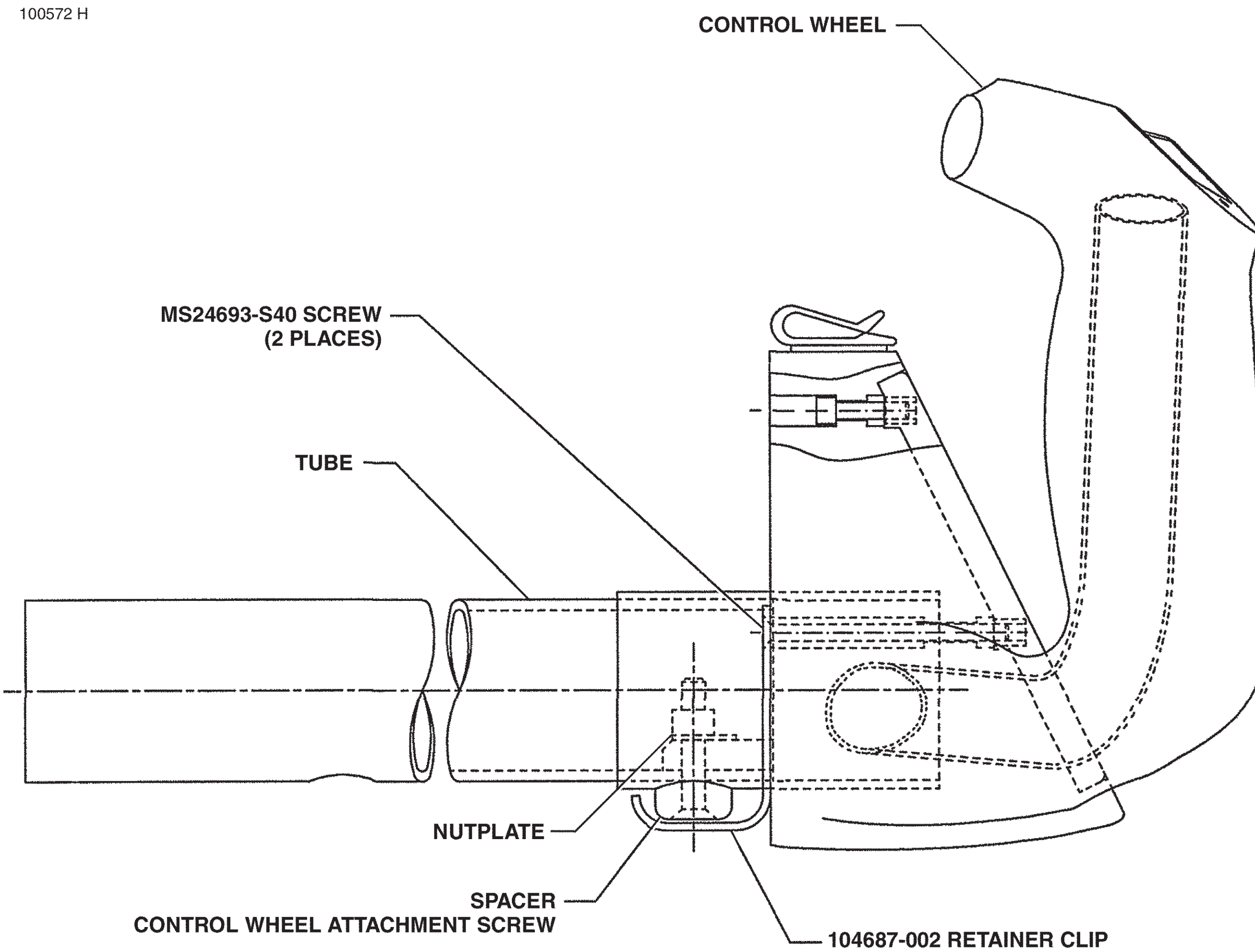
         (b) Slide control wheel onto tube.

         (c) Position and secure control wheel attachment bolt, spacer, washer, and locknut.

         (d) Position cover/pad assembly and secure with two screws from bottom of control wheel.

CHART 1
TROUBLESHOOTING AILERON CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between control wheel and aileron.	Cable tension too low.	Adjust cable tension.
	Linkage or torque rollers loose or worn.	Check linkage or torque rollers and tighten or replace.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
Resistance to control wheel rotation.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Control column horizontal cable improperly adjusted.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
	Bent aileron and/or hinge.	Repair or replace aileron and/or hinge.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Control wheels not synchronized.	Incorrect control column rigging.	Re-rig control column.
Control wheels not horizontal when ailerons are neutral.	Incorrect rigging of aileron system.	Re-rig aileron system.
Incorrect aileron travel.	Aileron stops not adjusted properly.	Adjust stops.
Correct aileron travel cannot be obtained by adjusting stops.	Incorrect rigging of aileron cables, and/or control wheel.	Re-rig controls.
Control wheel stops before control surfaces reach full travel.	Incorrect rigging between control wheel and control cables.	Re-rig controls.



Control Wheel Installation
Figure 1

C. S/N's 4636132 and up and in S/N's 4692001 and up.

The retainer clip (P/N 104687-002) and MS24693-S40 screws (2 ea.) are factory installed in S/N's 4636345 and up and in S/N's 4692001 and up. S/N's 4636132 thru 4636344 must individually procure and retrofit the retainer clip and screws. See "Figure 1" on page 27103.

(1) Removal

(a) Remove two (2) screws and retainer clip.

(b) Remove control wheel attachment screw and spacer.

(c) Slide control wheel off of tube.

(d) Disconnect control wheel switches wiring harness.

(2) Installation

(a) Degrease aft end of tube and inside of control wheel using acetone or naptha. Allow to dry.

(b) Degrease control wheel attachment screw, the spacer, and the nutplate (inside the tube) using acetone or naptha. Allow to dry.

(c) Prime inside of control wheel with Loctite 7649 (Piper P/N 279-073). Allow to dry.

(d) Connect control wheel switches wiring harness.

(e) Install control wheel onto tube using Loctite 271 (Piper P/N 279-128). Take care to ensure screw hole in control wheel aligns with screw hole in tube.

NOTE: This step must be accomplished promptly due to short cure time.

(f) Prime threads of control wheel attachment screw with Loctite 7649 (Piper P/N 279-073). Allow to dry.

(g) Install control wheel attachment screw (with spacer) into nutplate using Loctite 271 (Piper P/N 279-128).

NOTE: This step must be accomplished promptly due to short cure time.

(h) Position retainer clip to capture spacer and control wheel attachment screw. Secure with screws (2), being careful not to over tighten the screws and damage the logo medallion.

3. Control Column

See "Figure 3" on page 27106.

A. S/N's 4636001 thru 4636020 only.

(1) Removal

(a) Control wheel and tube:

1) Remove the four cap screws securing the tube and control wheel to the guide assembly. Note the relation of the tube to the guide.

2) Pull the wheel and tube out from the instrument panel.

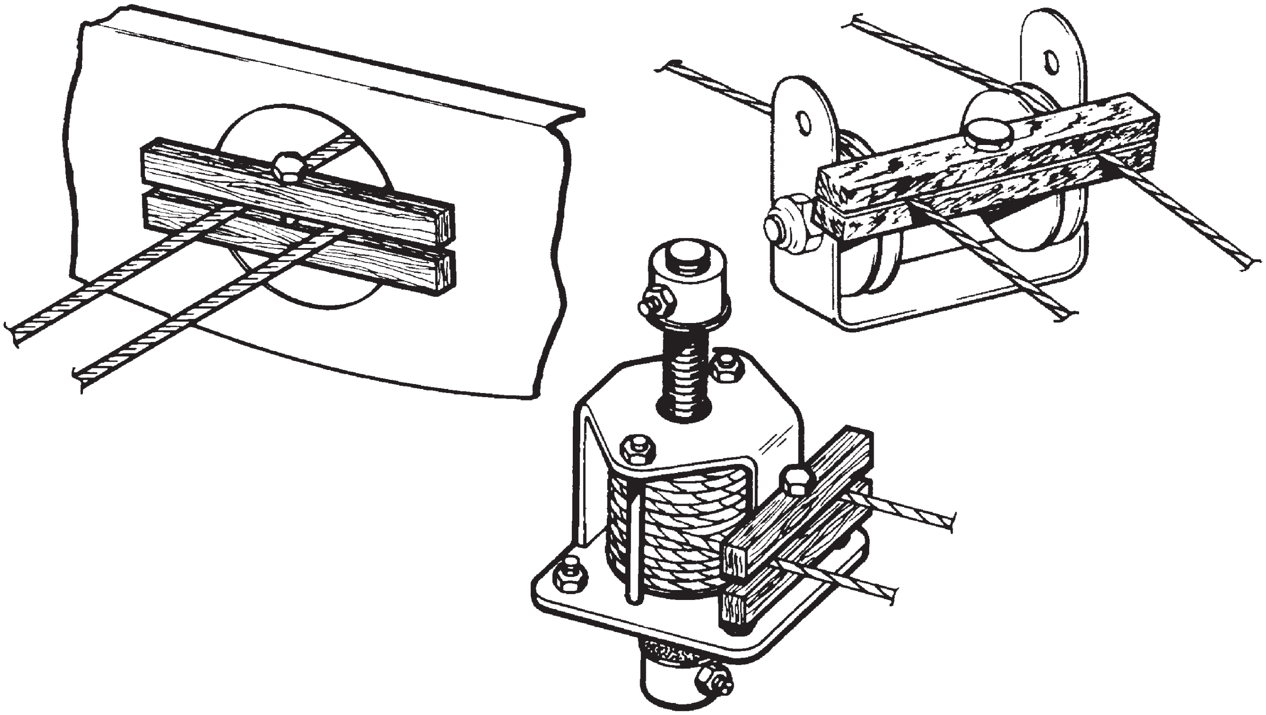
3) If not already done, disconnect control wheel switches wiring harness.

(b) Forward control column assembly:

CAUTION: LOOSEN BALANCE CABLES BEFORE REMOVING CONTROL COLUMN, TO PREVENT STRUCTURAL DAMAGE.

1) Block (see "Figure 2") the aileron primary cables at the pulleys mounted on the pressure bulkhead (F.S. 100.00) prior to loosening the balance cable between the control wheels.

2) Loosen the clamps securing the protective tube over the control wheel balance cable turnbuckle. Slide the tube to one side and loosen the turnbuckle to relieve tension on the cables.



Cable Blocking Methods
Figure 2

3) Remove the screws securing the cable retainers, both primary and balance, to the aileron quadrant at the forward end of the control columns.

4) Remove the nut and bolt securing the arms to the elevator torque tube behind the instrument panel.

5) Disconnect the control wheel switches electrical harness (if installed).

6) Remove the bolt, washers, and nut securing the control column tube to the quadrant shaft and separate the tube from the shaft. Make a note of washer buildup for reinstallation.

NOTE: The control column stop assy is fitted over the tube, secured by the same bolt, and will be removed with the tube.

7) The control column can now be removed from under the instrument panel.

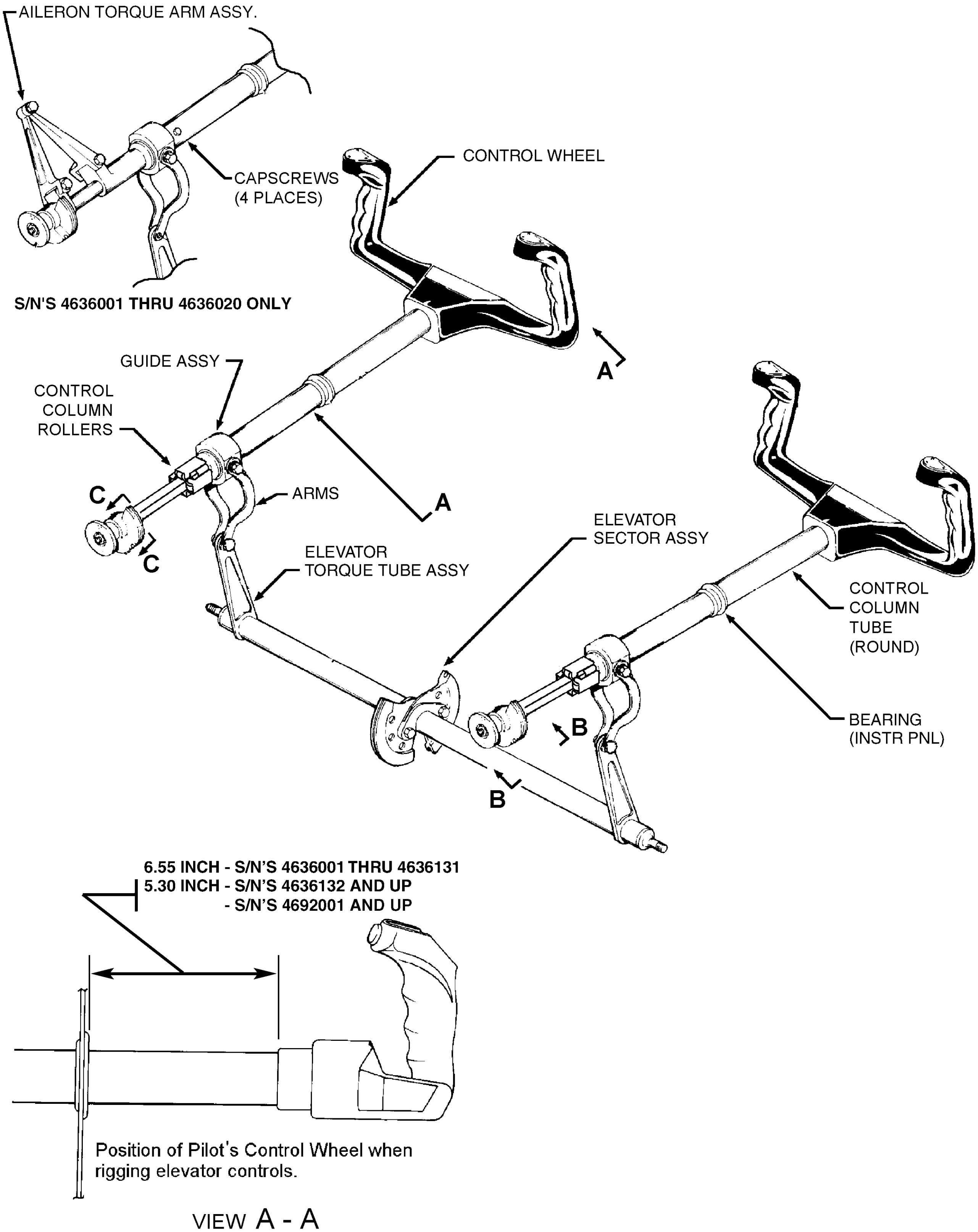
(2) Installation

(a) Forward control column assembly:

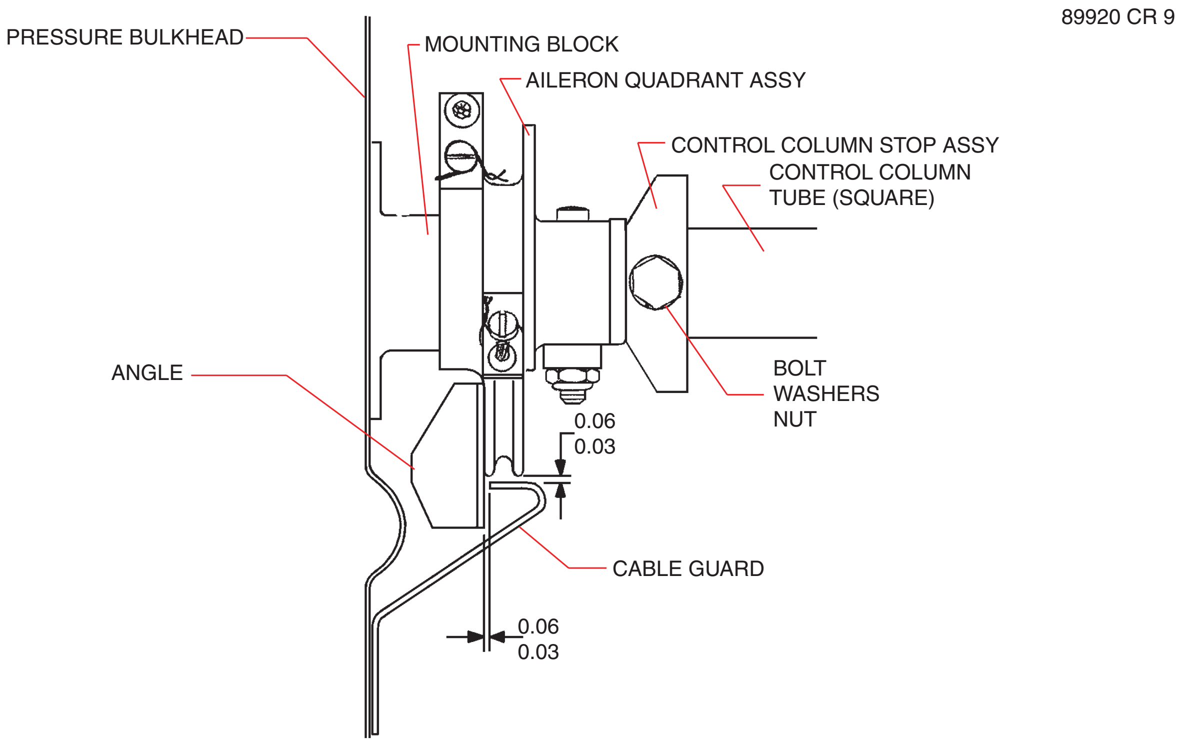
NOTE: First, ensure that the nut attaching control column rod to mount on forward end of control column is tightened to a torque of 40 to 45 inch-pounds.

1) Position the forward portion of the control column assembly in place behind the face of the instrument panel.

2) Ensuring the control column stop assembly is in place over the forward end of the control column tube, slide the tube over the quadrant shaft. Secure the control column tube and control column stop assembly to the quadrant shaft with bolt, washers, and nut.

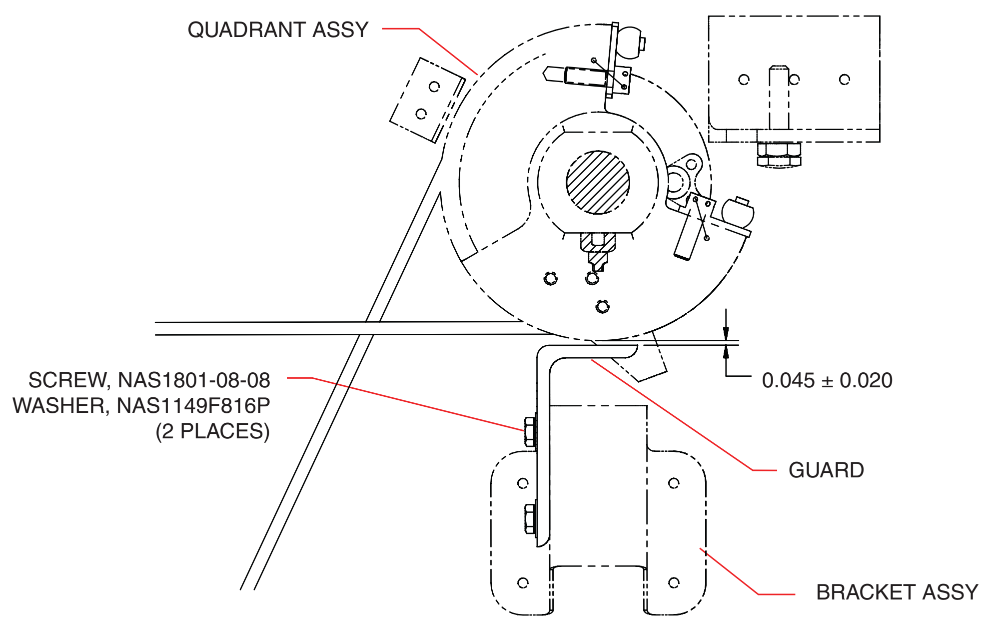


Control Column Installation
Figure 3 (Sheet 1 of 3)



S/N's 4636001–-4636687

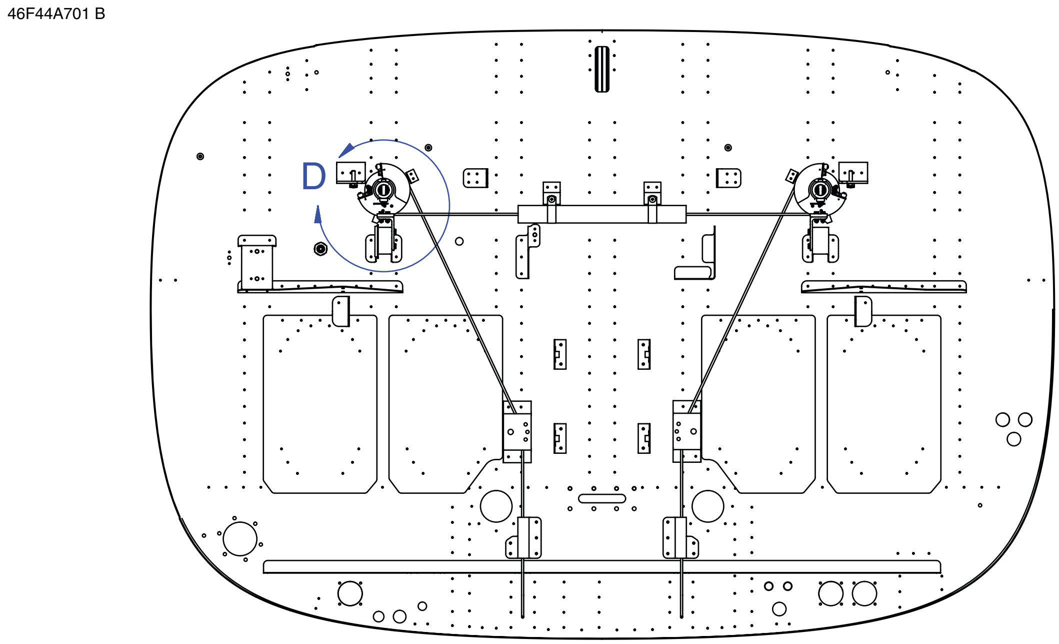
VIEW B - B
(LEFT SHOWN, RIGHT OPPOSITE)



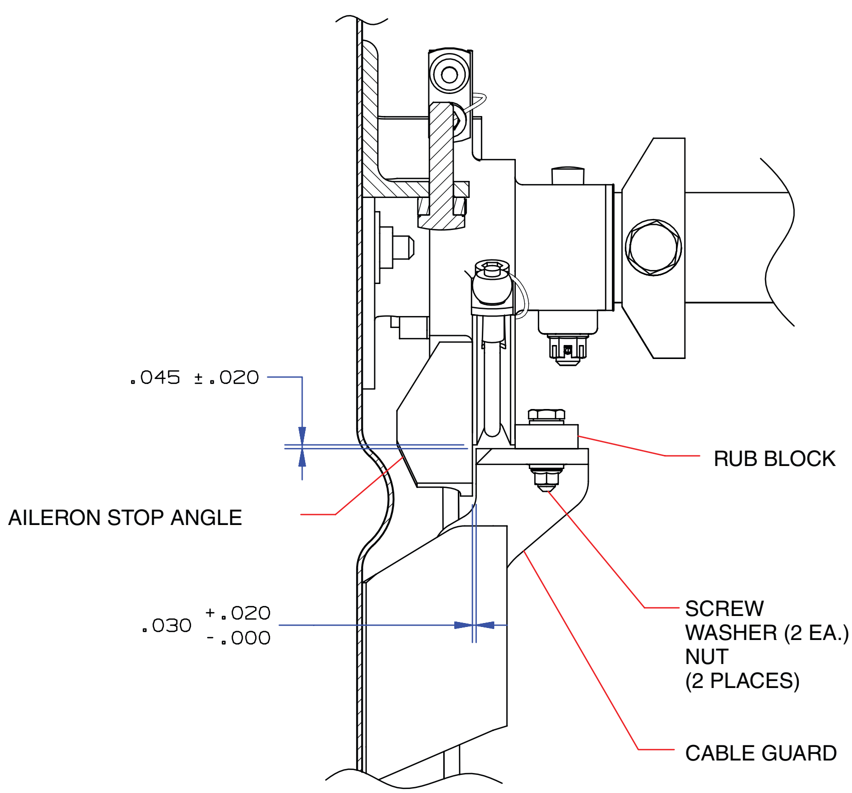
S/N's 4636688–4636724

VIEW C - C
(RIGHT SHOWN, LEFT OPPOSITE)

Control Column Installation
Figure 3 (Sheet 2 of 3)



LOOKING FORWARD AT PRESSURE BULKHEAD



S/N's 4636724 AND UP

DETAIL D

Control Column Installation
Figure 3 (Sheet 3 of 3)

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3) Attach the arms on the control column to the arm on the elevator torque tube. (The bolt beads should point outboard.)

4) Reconnect the control wheel switches electrical harness (if installed.)

5) Attach the primary and control wheel balance cables to the aileron quadrant at the forward end of the control column assembly and secure each cable end retainer (i.e. - lockplate) with a screw (see note below). Safety the cable end retainer screws with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

6) Remove the aileron primary cable blocks from the pulleys at F.S. 100.00.

7) Rig the aileron control system in accordance with instructions given in this chapter. Ensure that turnbuckles are safetied with locking clips and that protective tube is installed over turnbuckle on the control wheel balance cable.

(b) Control wheel and tube:

1) Connect the control wheel switches wiring harness.

2) Slide the wheel and tube assembly into the instrument panel.

NOTE: If bearing on instrument panel is loose when inserting wheel and tube assembly, use LOC-498-50 or LOC-495-50 Super Bond adhesive to bond bearing onto instrument panel.

3) Secure the tube and control wheel to the guide assembly with the four cap screws. Safety the screws.

4) Rig the aileron control system in accordance with instructions given in this chapter.

(c) Check the system for proper operation and freedom from binding.

B. S/N's 4636021 and up and in S/N's 4692001 and up.

(1) Removal

(a) Control wheel:

Remove control wheel as described for the appropriate serial number range under Control Wheel, above.

(b) Control column assembly:

CAUTION: LOOSEN BALANCE CABLES BEFORE REMOVING CONTROL COLUMN, TO PREVENT STRUCTURAL DAMAGE.

1) Block (see "Figure 2" on page 27105) the aileron primary cables at the pulleys mounted on the pressure bulkhead (F.S. 100.00) prior to loosening the balance cable between the control wheels.

2) Loosen the clamps securing the protective tube over the control wheel balance cable turnbuckle. Slide the tube to one side and loosen the turnbuckle to relieve tension on the cables.

3) Remove the screws securing the cable retainers, both primary and balance, to the aileron quadrant at the forward end of the control columns.

4) Remove the nut and bolt securing the arms to the elevator torque tube behind the instrument panel.

5) Disconnect the control wheel switches electrical harness (if installed).

6) Remove the bolt, washers, and nut securing the control column tube to the quadrant shaft and separate the tube from the shaft. Make a note of washer buildup for reinstallation.

NOTE: The control column stop assy is fitted over the tube, secured by the same bolt, and will be removed with the tube.

7) The control column can now be removed by lifting the forward end and sliding the assembly forward until the tube clears the instrument panel.

(2) Installation

(a) Control column assembly:

NOTE: Check bearing on instrument panel for security. If it is loose, use LOC-498-50 or LOC-495-50 Super Bond adhesive to bond bearing onto instrument panel.

1) Place the control column assembly under and behind the instrument panel, keeping the forward end raised until the aft end of the tube can be inserted into the bearing in the instrument panel. Slide the assembly aft and lower the forward end to its installed position.

2) Ensuring the control column stop assembly is in place over the forward end of the control column tube, slide the tube over the quadrant shaft. Secure the control column tube and control column stop assembly to the quadrant shaft with bolt, washers, and nut.

3) Attach the arms on the control column to the arm on the elevator torque tube. (The bolt beads should point outboard.)

4) Reconnect the control wheel switches electrical harness (if installed.)

5) Attach the primary and control wheel balance cables to the aileron quadrant at the forward end of the control column assembly and secure each cable end retainer (i.e. - lockplate) with a screw (see note below). Safety the cable end retainer screws with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

6) Remove the aileron primary cable blocks from the pulleys at F.S. 100.00.

(b) Control wheel:

Install control wheel as described for the appropriate serial number range under Control Wheel, above.

(c) Rig the aileron control system in accordance with instructions given in this chapter. Ensure that turnbuckles are safetied with locking clips and that protective tube is installed over turnbuckle on the control wheel balance cable.

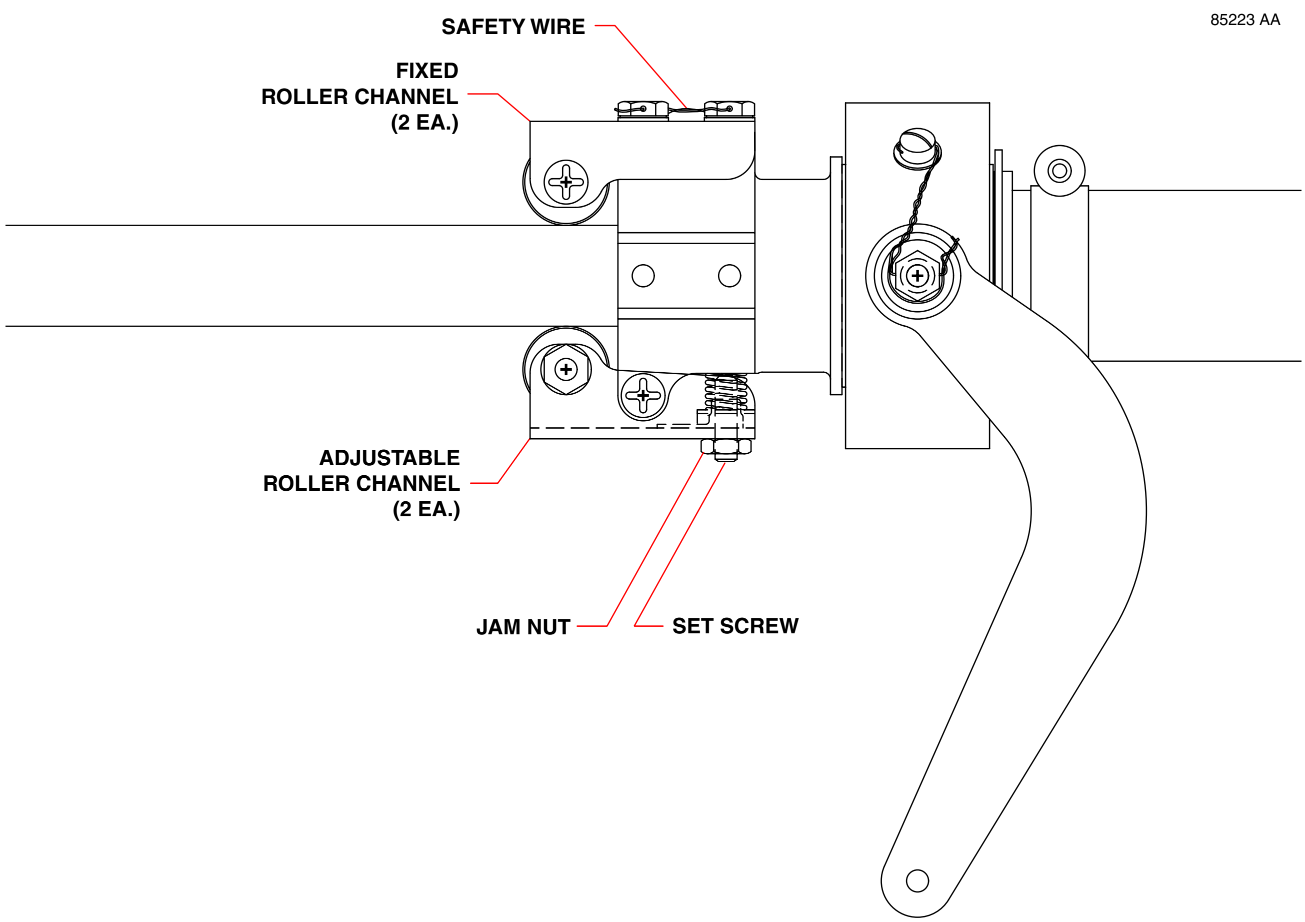
(d) Check the system for proper operation and freedom from binding.

(3) Control Column Rollers Adjustment (PIR-PPS50056-3, Rev. New.)

If aileron control freeplay becomes excessive, control column fore and aft movement binds or drags, or control wheel/control column up and down movement becomes excessive, the following procedure can be used to adjust the control column rollers (see “Figure 3” on page 27106 and “Figure 4”.)

(a) Required Equipment

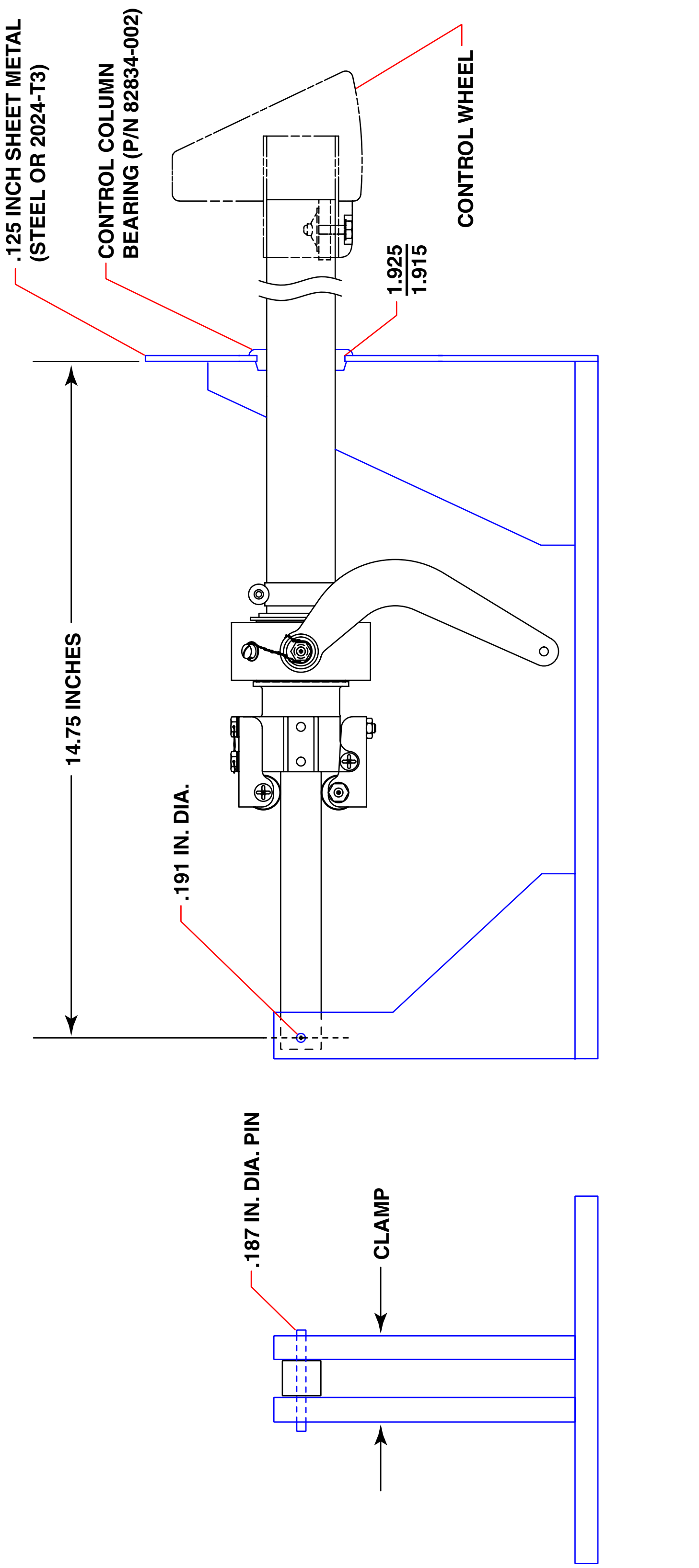
A holding fixture fabricated as shown in “Figure 5” on page 271014 is required for proper adjustment and inspection of the control column rollers. Critical dimensions/materials are shown. Other dimensions/materials are at the builders discretion providing the finished fixture is in the general configuration shown and sufficiently robust to support the control wheel / control column assembly with no flexing or deformation.



Control Column Rollers Adjustment
Figure 4

(b) Procedure

1) Remove control column per Removal, above.
2) Install control column assembly in holding fixture as shown. Clamp sides of fixture to prevent any rotation of square tube.
3) Temporarily install a control wheel on the round shaft end of column as shown using suitable hardware. Tighten attachment so that no play exists between the wheel and shaft.
4) Loosen the jam nuts and set screws in the two (2) adjustable roller channels and ensure that the channels are free to pivot. Adjust if necessary.
5) Tighten the four (4) bolts securing the two (2) fixed roller channels. Torque per 91-10-00, Chart 2. Do not safety at this time.
6) Slowly tighten the set screw in one of the adjustable roller channels while moving the control column in and out through the travel allowed by the holding fixture. Stop tightening when an increase in friction is felt. Back off the set screw approximately 1/8 turn and tighten jam nut.
7) Repeat step (e) above on the other adjustable roller channel.



Control Column Holding Fixture
Figure 5

8) Hold the fixture firmly in place on a flat smooth surface. Ensure that there is no play between the fixture and square tube end of control column. Ensure that there is no play between the control wheel and control column.

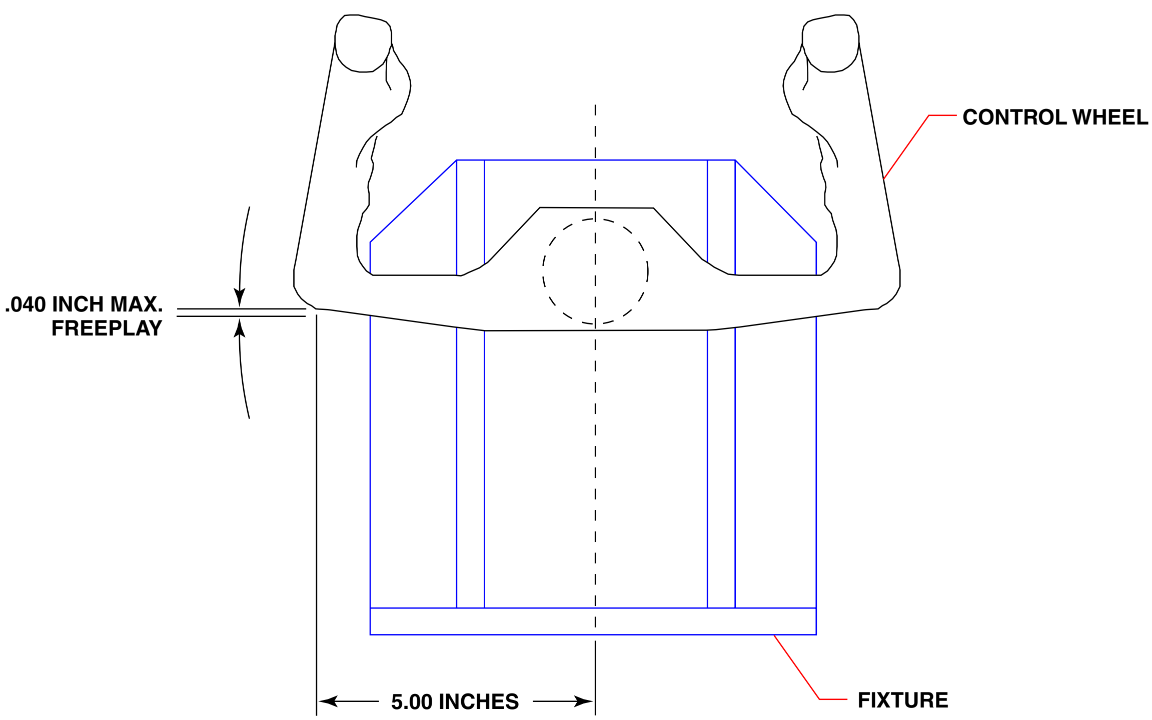
9) Using a dial indicator positioned five inches from the control wheel center, measure the lost motion between the rollers and the square tube by applying light rotational pressure to the control wheel. (Only enough pressure to move the wheel through the actual freeplay). Maximum allowable freeplay is shown in “Figure 6”.

NOTE: Lost motion may be reduced by switching the locations of roller channels on the roller channel support.

10) After adjustments are complete:
   a) Ensure all fasteners are properly torqued.
   b) Safety wire four (4) bolts securing fixed roller channels.
   c) Tighten and inspect each jam nut.

11) Remove control column assembly from holding fixture.

12) Install control column per Installation, above.



Control Column Maximum Freeplay
Figure 6

4. Aileron Control Cables

See “Figure 7”.

A. Removal

(1) For removal of any control cables in the fuselage or wings, first determine which cables are to be removed and then proceed with the appropriate set of instructions which follow.

(2) To remove either right or left primary control cables that are located in the fuselage, the following procedures may be used.

(a) Remove access panels from the cabin floor at fuselage station 171.43. This will allow you access to either turnbuckle.

(b) Separate the primary control cable at the turnbuckles.

(c) Remove the access panels from the cockpit floor and then remove pulley cable guards from pulleys located in front of the pressure bulkhead.

(d) Disconnect the primary cables from the control column quadrant assemblies.

(e) Draw the cables back through the floor.

(3) To remove the primary control cable in either wing, proceed as follows:

(a) Remove the access panel from the cabin floor at fuselage station 171.43.

(b) Separate the primary cable at the turnbuckle within the fuselage and disconnect the opposite end at the sector out in the wing located aft of the rear spar at wing station 175.498.

(c) Remove the cable guards at the pulleys located along the aft side of the wing rear spar.

(d) Remove the control cable from the fuselage by removing the retaining rings on the fin type air seal and pulling the cable from the fuselage.

(4) Either balance cable may be removed by the following procedure:

(a) Remove the access panels from the cabin floor.

(b) Separate the balance cables at the turnbuckle located below the right side of the cabin floor at F.S. 171.43.

(c) Disconnect the balance cable from the sector in the wing located at wing station 175.498 along the aft side of the rear spar.

(d) Remove the cable guards at the pulleys along the wing rear spar.

(e) Remove the control cable from the fuselage by removing the retaining rings on the fin type air seal and pulling the cable from the fuselage.

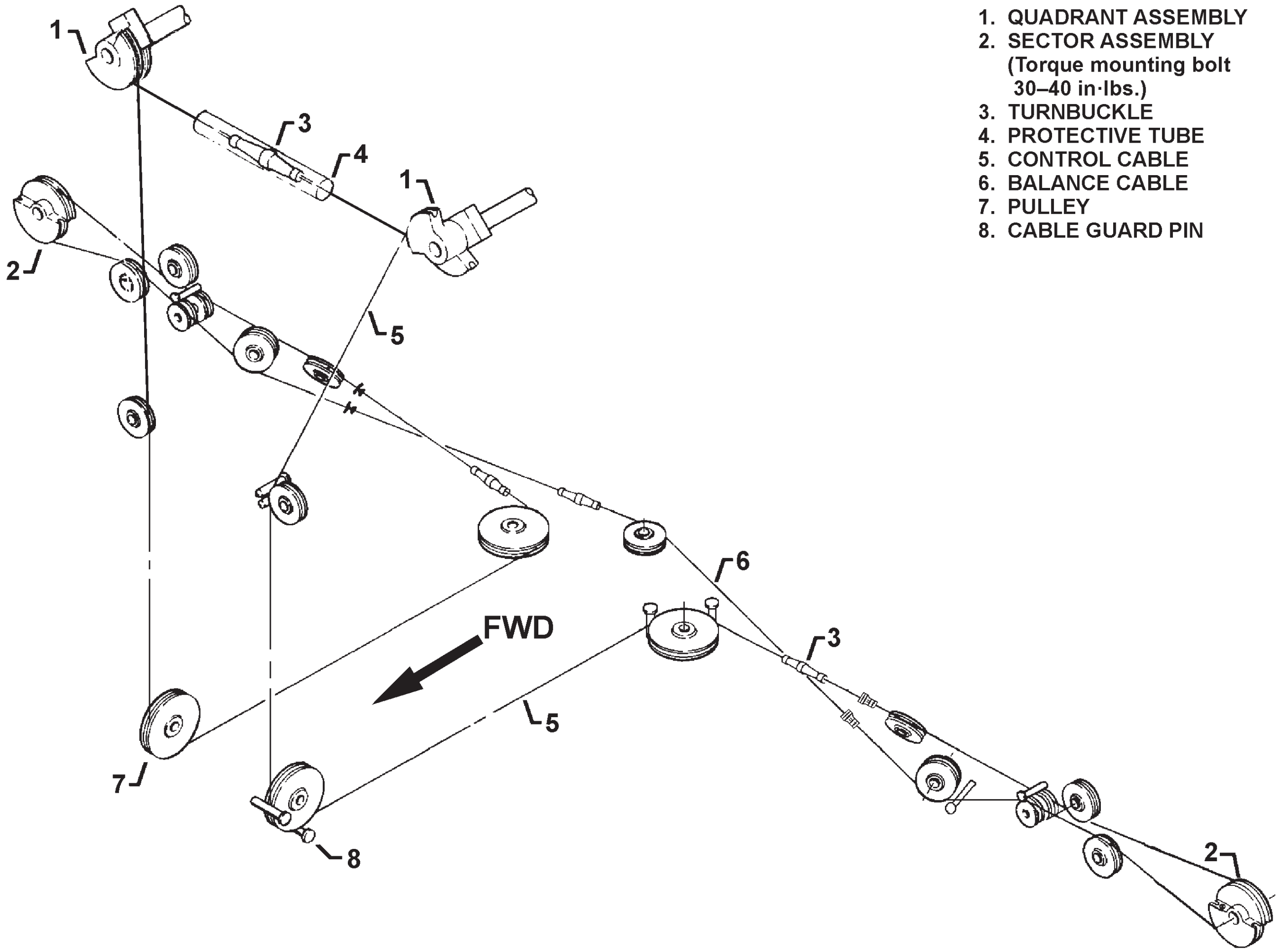
B. Installation

(1) The installation of either the right or left primary control cable, located in the fuselage may be accomplished as follows:

(a) Draw the cable through the floor, and around the pulleys at the forward pressure bulkhead.

(b) Connect the cable end to the quadrant assembly at the forward end of the control column and secure the cable end retainer (i.e. - lockplate) with a screw (see note below). Safety the cable end retainer screw with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.



Aileron Controls
Figure 7

(c) Ensure that the cable is properly placed around the pulley and install cable guards.

(d) Route cable around the pulley at fuselage station 171.43 and install cable guard.

(e) Connect cable turnbuckle to primary cable end from wing, and ensure proper rigging of system.

(f) Install all access panels removed.

(2) The primary control cable in either wing may be installed by the following procedure:

NOTE: Make sure that fairlead P/N 89637-003 attaching nuts and washers are installed against the aft wing spar just outboard of flap actuator push rod cutout, not on fairlead.

(a) Draw the central cable along the wing rear spar, ensuring proper routing around the pulleys.

(b) Connect the end of the primary cable to the aileron sector at wing station 175.498 and secure the cable end retainer (i.e. - lockplate) with a screw (see note below). Safety the cable end retainer screw with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

(c) Install the fin-type cable air seal on the cable and install the seal in the bulkhead.

(d) Route the cable through the fuselage.

(e) Connect the turnbuckle to the primary cable within the fuselage, and ensure proper rigging of system.

(f) Install all access panels removed.

(3) Either balance cable may be installed by the following procedure:

(a) Connect the cable end to the sector in the wing at wing station 175.498 and secure the cable end retainer (i.e. - lockplate) with a screw (see note below). Safety the cable end retainer screw with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

(b) Route the cable along the rear spar, ensuring its proper routing through the pulleys.

(c) Route the cable through the fuselage and then place the retainer rings on the seal. (Refer to ”Figure 5” on page 270011.)

(d) Connect the balance cable turnbuckle ends together and rig the ailerons as required. Check complete system for operation, see Caution under Rigging and Adjustment.

(e) Install all access panels removed.

C. Rigging and Adjustment

(1) Lock the control wheels together in the aileron neutral position using a suitable tool.

(2) Lock the ailerons in neutral position using a suitable contour fixture at the inboard ends of the ailerons. Neutral position is defined by wing loft contour at this location.

(3) Adjust the aileron control cables to the tension specified in “Chart 2” on page 27004.

(4) Remove the locking fixtures. Aileron neutral position should be maintained at 0 + 1, - 0 degree with the control wheels in neutral.

(5) Use the pilot’s control wheel to move the ailerons and adjust the aileron travels at the primary stops (at the aileron drive sectors on the wing spars) to give the ailerons up and down movement specified in “Chart 2” on page 27004. Total freeplay measured at the aileron inboard trailing edge shall not exceed that specified in “Chart 2” on page 27004.

(6) Use the pilot’s control wheel to hold the ailerons firmly against the primary stops. Adjust the secondary aileron stops (on the forward pressure bulkhead) to maintain a cushion (gap) of 0.09 ± 0.03 inch in each direction.

(7) Check system for proper operation.

(8) After final aileron cable rigging, ensure clearance dimensions between quadrant assembly, angle and cable guard are maintained with full range of rotational movement of the quadrant assemblies at both left hand and right hand control column assemblies. Hand form the cable guard, as needed, to achieve clearances shown in “Figure 3” on page 27106.

(9) Check for proper operation per “Post-Rigging Inspection”, below.

D. Post-Rigging Inspection

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF AILERONS. UPON COMPLETION OF AILERON RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RIGHT AILERON TRAILING EDGE MOVES UP AND THE LEFT AILERON TRAILING EDGE MOVES DOWN WHEN THE CONTROL WHEEL IS TURNED RIGHT; AND THAT THE LEFT AILERON TRAILING EDGE MOVES UP AND THE RIGHT AILERON TRAILING EDGE MOVES DOWN WHEN THE CONTROL WHEEL IS TURNED LEFT.

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any aileron control cable replacement, disconnect, or rigging perform the following post-rigging inspection:

(1) Rotate the pilot’s control wheel CLOCKWISE, as if to bank the airplane to the RIGHT, until reaching the primary stop. Visually confirm the following:

(a) The RIGHT hand wing aileron is deflected UP. The trailing edge of the aileron will be noticeably and significantly ABOVE the trailing edge of the RIGHT wing at the far RIGHT end of the aileron, near the wing tip.

(b) The LEFT hand wing aileron is deflected DOWN. The trailing edge of the aileron has moved noticeably and significantly BELOW the trailing edge of the LEFT wing at the far LEFT end of the aileron, near the wing tip.

(2) Rotate the pilot’s control wheel COUNTERCLOCKWISE, as if to bank the airplane to the LEFT, until reaching the primary stop. Visually confirm the following:

(c) The LEFT hand wing aileron is deflected UP. The trailing edge of the aileron has moved noticeably and significantly ABOVE the trailing edge of the LEFT wing at the far LEFT end of the aileron, near the wing tip.

(d) The RIGHT hand wing aileron is deflected DOWN. The trailing edge of the aileron has moved noticeably and significantly BELOW the trailing edge of the RIGHT wing at the far RIGHT end of the aileron, near the wing tip.

(3) Repeat the above steps for the copilot controls.

(4) Make a logbook entry documenting completion of this inspection.

5. Aileron Tab

A fixed trim tab is located on the trailing edge of the right aileron. The tab is ground adjustable only. Adjust tab as needed within limits shown in “Figure 8”.

6. Aileron - Rudder Interconnect

See “Figure 9” on page 271021.

The aileron - rudder interconnect installation is located underneath the floor between the frames at F.S. 131.84 and F.S. 139.80. The aileron - rudder interconnect is bolted to the left aileron cable. The aileron - rudder interconnect guide is bolted to the left rudder cable.

The aileron-rudder interconnect is rigged and adjusted as follows:

NOTE: All aileron and rudder rigging must be completed prior to rigging the aileron - rudder interconnect.

A. Gain access to the aileron rudder interconnect.

B. Lock the aileron and rudder systems in neutral (with the control wheels and rudder pedals neutral, the rudder is 2° right of 0°).

C. Ensure that the aft cable clamp on the left rudder cable is positioned 3.15 inches from the aft face of the F.S. 139.80 fuselage frame.

NOTE: Dimensions are provided for initial set-up reference only and are not an inspection requirement. Actual position of clamps aft of STA 139.80 frame shall be adjusted as required to ensure positive clearance to frames and cable guides thru full range of travel, provided dimensions to STA 131.84 frame are maintained.

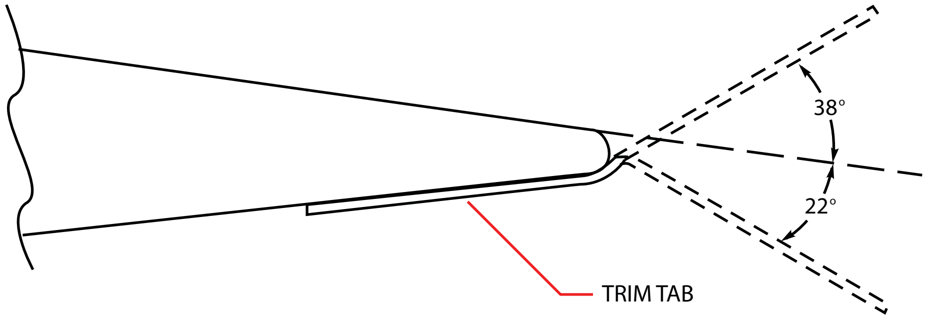
D. Ensure that the aft cable clamp on the left aileron cable is positioned 2.00 inches from the aft face of the F.S. 139.80 fuselage frame.

E. Ensure that the forward side of the forward arm of the aileron - rudder interconnect is positioned 1.70 inches from the aft face of the F.S. 131.84 fuselage frame.

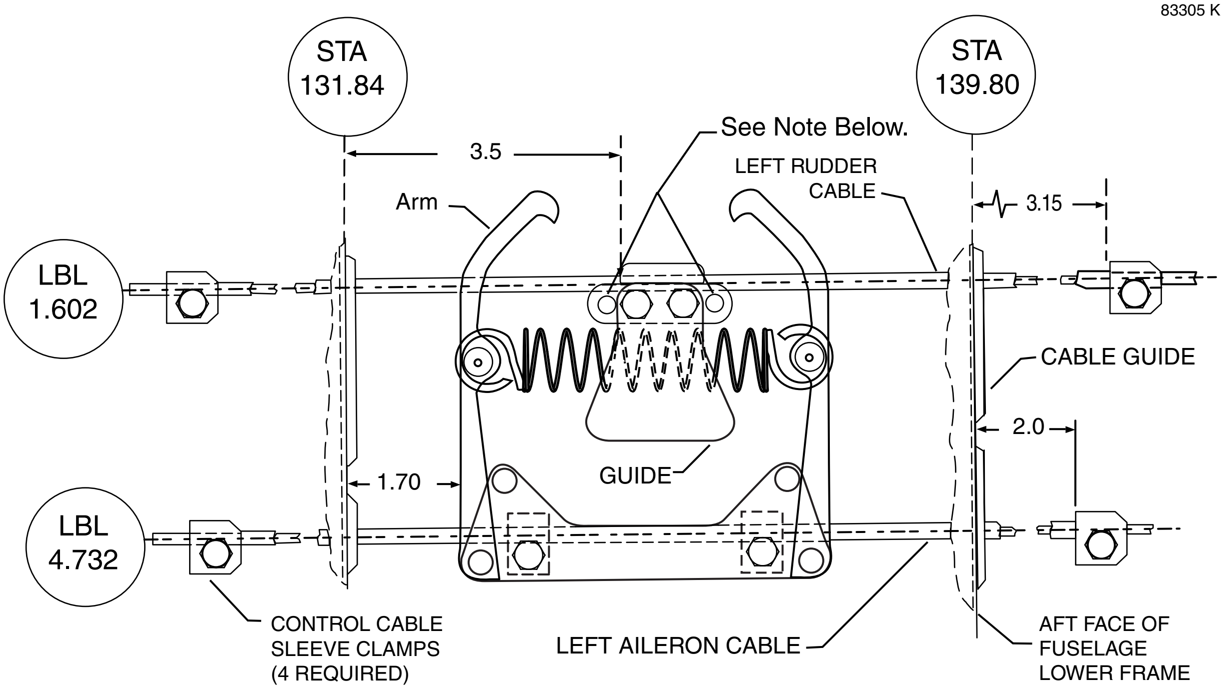
F. Ensure that the forward side of the guide (on the left rudder cable) is positioned 3.50 inches from the aft side of the F.S. 131.84 fuselage frame.

WARNING: AFTER RIGGING THE INTERCONNECT, DEMONSTRATE THAT BOTH THE AILERONS AND THE RUDDER CAN OBTAIN FULL TRAVEL AND BE DEFLECTED TO THEIR RESPECTIVE STOPS WITHOUT INTERFERING WITH EITHER FRAME F.S. 131.84 OR FRAME F.S. 139.80. WHEN DEMONSTRATING THIS, MOVE BOTH THE AILERON AND RUDDER IN THE SAME DIRECTION TO THEIR RESPECTIVE STOPS AND THEN HOLD FIRMLY IN POSITION WHILE THE FRAMES ARE INSPECTED FOR INTERFERENCE WITH THE INTERCONNECT.

G. After adjustment, verify smooth, free operation by watching the aileron - rudder interconnect while an assistant cycles the yoke and pedals through full control movement.



Aileron Trim Tab Adjustment Limits.
Figure 8



NOTE: In early production airplanes, these pins may interfere with the F.S. 139.80 frame. If so, reverse the pins and add two (2) NAS 1149FN416P washers between the cotter pin and the plate.

Aileron - Rudder Interconnect Installation
Figure 9

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RUDDER AND TAB

1. Troubleshooting

   See “Chart 1”.

CHART 1
TROUBLESHOOTING RUDDER CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between rudder pedals and rudder.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Bolts attaching forward or aft sectors are loose.	Tighten sector bolts.
Excessive resistance to rudder pedal movement.	System not lubricated properly.	Lubricate system.
	Rudder pedal torque tube bearing in need of lubrication.	Lubricate torque tube bearings.
	Cable tension too high.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Incorrect rudder travel.	Rudder sector stop incorrectly adjusted.	Re-rig sector stops.
	Nose wheel contacts stops before rudder.	Re-rig nose wheel stops.
Trim control knob moves with excessive resistance.	System not lubricated properly.	Lubricate system.

2. Rudder Pedal Assembly

To change bearings on rudder bar assembly, follow these steps:

A. Remove nut and washers from end of rudder bar assembly.

B. Remove nuts, bolts and washers that secure the bearing block to the tube support bracket.

C. Remove bearing.

D. Replace bearing using the shrink fit process. Bond the bearing with Loctite 290.

E. Securing bearing block to tube support bracket with nuts, bolts and washers.

F. Place two washers over the threaded end of the rudder bar and secure with the nut.

3. Rudder Controls

A. Removal

(1) To remove the rudder torque tube sectors:

(a) Loosen cables.

(b) Remove the screw which secures the rudder cable end retainer (i.e. - lockplate) to the sector.

(c) Remove the nuts, bolts and washers that attach the sector to the rudder torque tube.

(d) Remove the sector.

(2) To remove the forward rudder control cables.

(a) Remove the screw which secures the rudder cable end retainer (i.e. - lockplate) to the rudder bar sector. Remove the cable from the sector.

(b) Remove the tailcone access panel.

(c) Disconnect the rudder turnbuckles by the rudder tube sector in the aft section of the fuselage.

(d) At the aft pressure bulkhead, pull the control cable through the fin type air seal. (Refer to ”Figure 5” on page 270011.)

(e) Pull the cable aft and remove it from the airplane.

(3) To remove the aft rudder cable.

(a) Remove the screws and washers that secure the two cable guard plates at both sides of the sector. Remove the safety plates.

(b) Disconnect the turnbuckles if not previously accomplished.

(c) Remove the cotter pins from the rear of the sector and remove the aft cable from the sector.

(4) To remove the rudder torque tube sector:

(a) Remove the aft rudder cable if not previously accomplished.

(b) Remove the nuts, bolts and washers that attach the rudder sector to the rudder torque tube.

(c) Remove the rudder sector.

B. Installation

(1) To install the rudder torque tube sector:

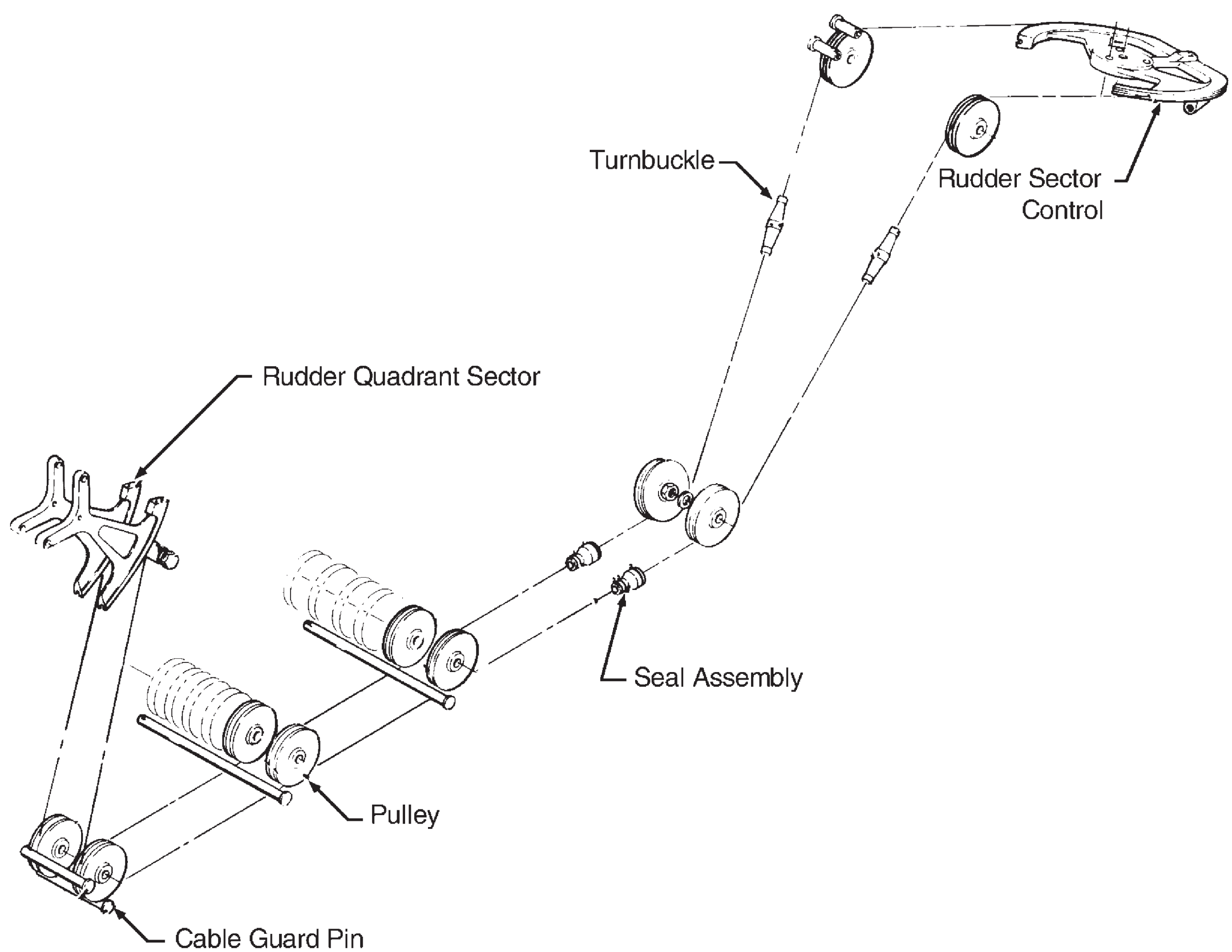
(a) Position the sector on the torque tube.

(b) Attach the sector to the torque tube with nuts, bolts and washers.

(2) To install the rudder bar sector:

(a) Position the sector on the rudder bar assembly.

(b) Secure the sector to the rudder bar assembly with nuts, bolts and washers.



Rudder Controls
Figure 1

(3) To install the aft rudder cable:

(a) Position the cable on the sector.

(b) Install the two guard plates with screws and washers. Replace the cotter pins in the rear of the sector.

(c) Attach the aft cable to the forward cables at the turnbuckles. Safety the turnbuckles with locking clips.

(4) To install the forward rudder cable:

(a) Push the cable through the fin type air seal. (See ”Figure 5” on page 270011.) Thread the cable forward through the airplane.

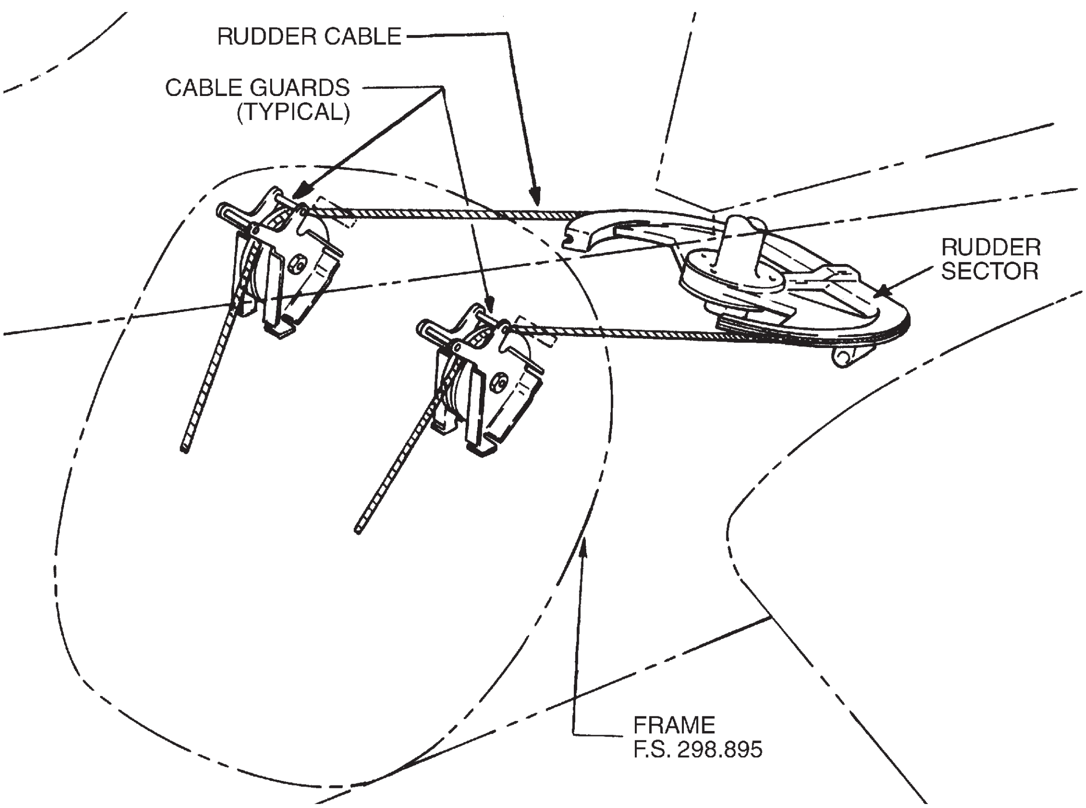
(b) Connect the forward and aft rudder cables at the turnbuckles in the aft end of the airplane.

(c) Attach the forward ends of the forward cables to the rudder bar sector and secure the cable end retainer (i.e. - lockplate) with a screw (see note below). Safety the cable end retainer screw with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

(5) Ensure that all pulley cable guards have been properly installed (i.e. - with the cables routed inside/under the cable guards - see “Figure 2”).

(6) Set the rudder cable tension and check the rigging per instruction in this chapter.



Cable Guard Installation (Typical)
Figure 2

(7) After setting the cable tension, check all cable ball ends for proper seating in the retainers.

(8) Check the system for proper operation. See the Caution under Rigging and Adjustment.

(9) Rig the rudder control system.

C. Rigging and Adjustment

(1) Use a suitable tool to check the rudder travel for its proper travel (perpendicular to the hinge line) left and right from the zero degree position (streamlined with the vertical stabilizer). The left travel should be as specified in “Chart 2” on page 27004 and the right travel should be as specified in “Chart 2” on page 27004. Adjust the stops on the rudder sector as required.

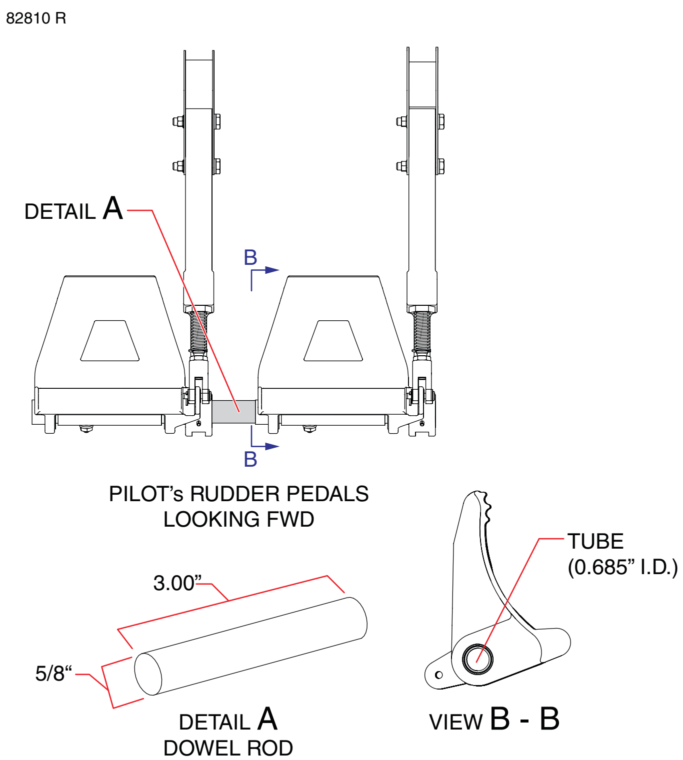
(2) Move rudder 2 degrees right of the zero degrees (streamlined) position. Lock rudder in this position.

WARNING: THE DOWEL ROD IN THE FOLLOWING STEP RESTRICTS FREE AND CORRECT MOVEMENT OF RUDDER PEDALS. THE DOWEL ROD MUST BE REMOVED ONCE ALIGNMENT OF THE NOSE GEAR ASSEMBLY IS COMPLETED.

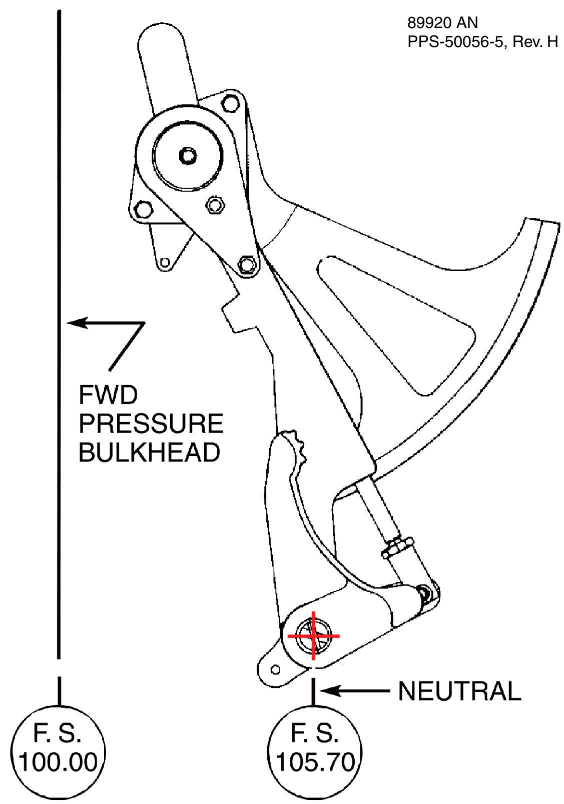
(3) Align rudder pedals by inserting a dowel inside the right rudder pedal tube and then withdrawing it into the left rudder pedal tube until it stops. Set pedals at F.S. 105.70 (see ”Figure 3”) and verify they remain aligned.

(4) Disconnect bungee from forward steering arm.

(5) Install a suitable tool which will lock the nose wheel in alignment with the longitudinal axis of the aircraft.



Rudder Pedals Aligned



Rudder Pedals at Neutral Position

Locking Rudder Pedals in Neutral
Figure 3

(6) Adjust the steering arm by loosening the steering arm mount and moving the adjuster on the bottom of the mount plate fore and aft (see 32-20-00, Figure 4). Place 0.020 inch thick feeler gauges (two required) between the steering horn rollers and the steering arm, and adjust steering arm to achieve contact between the steering horn roller, the feeler gauge, and steering arm, for both left and right hand sides simultaneously. When properly adjusted, the acceptable clearance between the rollers and the steering arm is between 0.010 and 0.030 inches. This clearance requirement applies to both rollers at the same time.

NOTE: Measure the gap with the steering arm parallel to the steering horn rollers. While measuring this clearance, the gear should be down and locked and the airplane's weight should be on the gear.

(7) Torque the steering arm mounting nut and the locking assembly jam nuts.

(8) Ensure that the two steering pushrods are holding the steering bellcrank perpendicular to B.L. 0.00 (refer to 32-50-00, Steering Pushrods Installation, if adjustment is necessary).

(9) Reconnect the steering bungee between the steering arm and the steering bellcrank. Adjust length as required so that there is no load on the bungee springs while the nose wheel is aligned with the longitudinal axis of the airplane. Tighten jam nuts.

(10) Adjust the rudder cable turnbuckles to obtain tension as specified in "Chart 2" on page 27004, with the rudder set at 2 degrees right of neutral. Adjust the cables evenly to avoid uneven strain on airplane components.

(11) Remove any centering tools (such as the rudder pedal dowel) that had been previously installed.

(12) Place the airplane on jacks (refer to 7-10-00).

(13) Depress each of the pilot's rudder pedals until the rudder sector firmly contacts its stops. Adjust the forward stops (on the bellcrank in the nose gear tunnel) to provide 0.06 to 0.09 inch clearance in each direction.

(14) Operate the rudder through its full travel range with the tab set at both full left and full right positions. Operation shall be smooth and free. With tab neutral and rudder held securely against each stop, determine free play of tab. Total free play, measured at tab trailing edge, shall not exceed 0.06 inches.

(15) Proceed to "Post-Rigging Inspection" on page 27206.

D. Post-Rigging Inspection

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF RUDDER. UPON COMPLETION OF RUDDER RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RUDDER TRAILING EDGE MOVES RIGHT WHEN THE RIGHT PEDAL IS DEPRESSED; AND THAT THE RUDDER TRAILING EDGE MOVES LEFT WHEN THE LEFT PEDAL IS DEPRESSED.

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

NOTE: This inspection may need to be performed with the airplane on jacks.

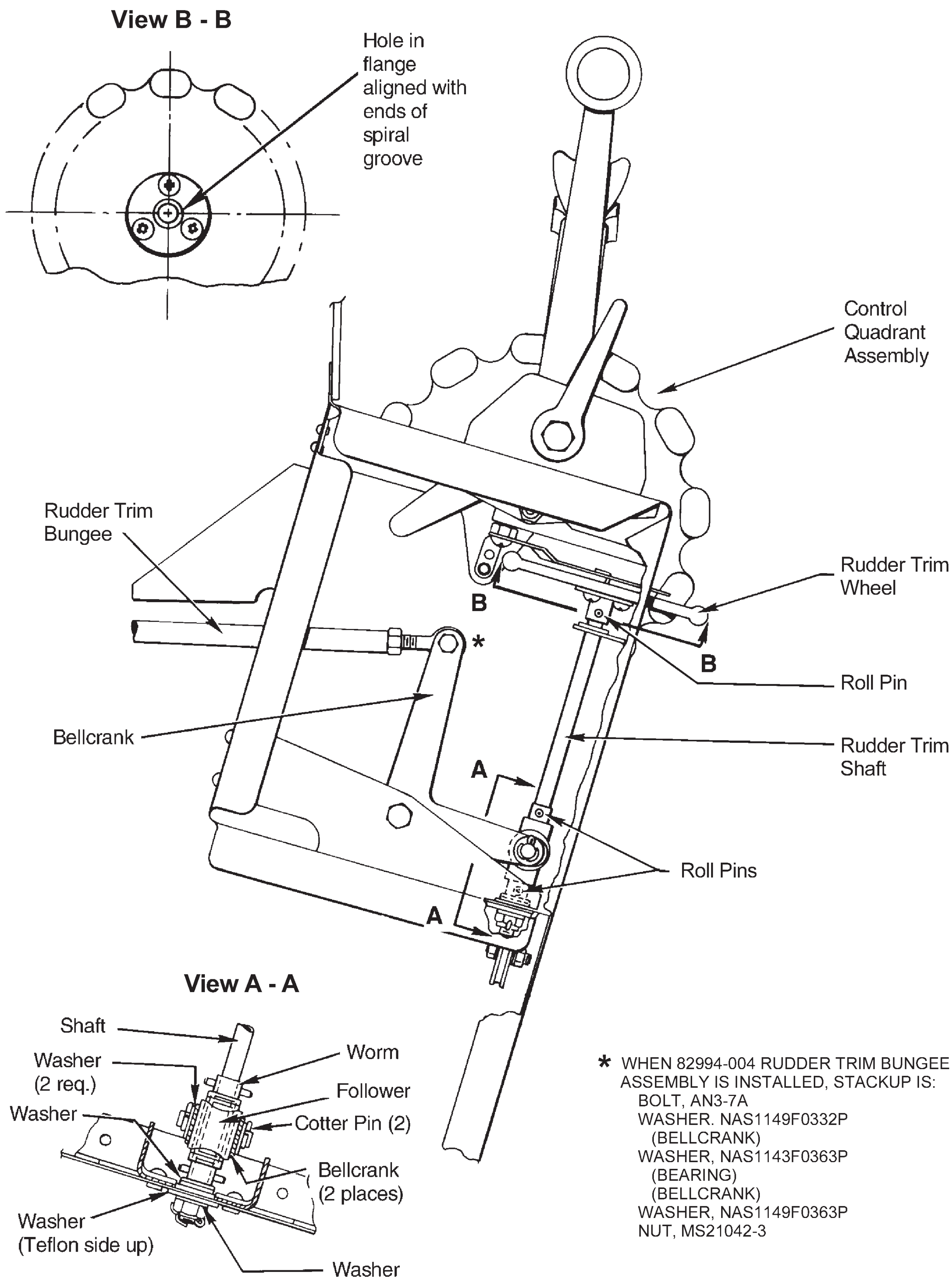
Upon completion of any rudder control cable replacement, disconnect, or rigging perform the following post-rigging inspection:

(1) Depress the pilot's RIGHT rudder pedal as if to yaw the airplane to the RIGHT, until reaching the primary stop. Visually confirm the trailing edge of the rudder will be noticeably and significantly RIGHT of the trailing edge of the vertical fin.

(2) Depress the pilot's LEFT rudder pedal as if to yaw the airplane to the LEFT, until reaching the primary stop. Visually confirm the trailing edge of the rudder will be noticeably and significantly LEFT of the trailing edge of the vertical fin.

(3) Repeat the above steps for the copilot controls.

(4) Make a logbook entry documenting completion of this inspection.



Rudder Trim Installation
Figure 4

4. Rudder Trim Controls

A. Removal

(1) To remove rudder trim controls in console:

(a) Remove access panel on left side of pedestal.

(b) Remove nut, washer and teflon-faced washer from bottom of rudder trim control shaft.

(c) Remove cotter pins and washers from the follower; remove the bellcrank arms from the arms on the follower.

(d) Rotate the trim wheel until roll pins in shaft point fore and aft. Drive roll pins out (three places).

(e) Remove the three screws which hold the wheel to the shaft.

(f) Slide the wheel aft through the wheel slot in the console.

(g) Slide the rudder trim control shaft down and out of the console.

(h) Remove the follower, worm and washer from inside the console.

(2) To remove the rudder trim bungee:

(a) Remove the nut, bolt and washers that secure rod end to bellcrank in console.

(b) Remove nut, bolt, cotter pin and washers that secure bungee housing to rudder bar assembly. Remove bungee from airplane.

(3) To disassemble rudder trim bungee:

(a) Note length of trim bungee.

(b) Cut safety wire and remove the two clips from the housing.

(c) Slide shaft assembly from housing.

B. Installation

(1) To assemble the rudder trim bungee assembly:

(a) Slide forward clip onto housing.

(b) Lubricate shaft at each end of spring with Aero Lubriplate (white) and insert shaft assembly into housing. Install aft clip.

(c) Check for 0.020 inch max. end play. Add 62833-127 washers as required between forward clip and end of spring on the shaft assembly to bring the end play into tolerances.

(d) Ensure that length of trim bungee is correct (as recorded in step (3), Removal, above).

(2) To install rudder trim bungee:

(a) Attach the housing end of the bungee to the rudder bar assembly. Secure with bolt, washers and nut. The thin washer goes under the nut. Safety the nut with a cotter pin.

(b) Attach the rod end of the bungee to the bellcrank. Safety the nut with a cotter pin.

(3) To install the rudder trim controls in the console:

(a) Slide the rudder trim control shaft partially up through the hole in the support bracket in the console.

(b) Place over the shaft, in order, the washer and worm. Screw the follower onto the worm. Slide the shaft all the way up into the upper hole.

(c) Align the holes in the worm and the holes in the shaft. Install roll pins in the hole.

(d) Place trim wheel flange on top of the shaft. Align holes in shaft and flange. Install roll pins.

(e) Slide wheel into slot on face of console. Align ends of spiral groove with roll pin hole in flange. Secure wheel to flange with screws and washers.

(f) On bottom of rudder trim shaft, place the teflon-faced washer (teflon side up) and washer.

(g) Install nut and washer on end of shaft. The shaft should rotate freely with minimum end play. Install the cotter pin.

(h) Attach bellcrank arms to follower with washers and cotter pins.

(i) Rig the rudder trim system per “Rigging and Adjustment”, below.

C. Rigging and Adjustment

(1) Ensure rudder rigging is correct per “Rigging and Adjustment” on page 27205.

(2) Using a suitable tool, set rudder at 3 degrees to left of 0.

(3) Disconnect bungee from bellcrank in pedestal.

(4) Mark rudder trim wheel and count revolutions for full travel, stop to stop.

(5) Set trim wheel and trim indicator at midpoint of trim wheel travel with rudder set 3 degrees to the left of 0.

(6) Adjust length of bungee rod as required to reconnect it with no load on the bungee springs.

NOTE: When 82994-004 Bungee Assembly is installed, safetywire with MS20995-C32.

(7) Check full range travel to ensure that indicator guide pin does not reach either end of spiral groove in trim wheel before the stop is engaged (at roll pin on worm).

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF RUDDER. UPON COMPLETION OF RUDDER TRIM RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RUDDER TRAILING EDGE MOVES LEFT WHEN THE RUDDER TRIM WHEEL IS TRIMMED LEFT; AND, THAT THE RUDDER TRAILING EDGE MOVES RIGHT WHEN THE RUDDER TRIM WHEEL IS TRIMMED RIGHT.

(8) Proceed to “Post-Rigging Inspection” on page 27209.

D. Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any rudder trim system component replacement, disconnect, or rigging perform the following post-rigging inspection:

(1) Set the rudder trim wheel in the neutral position.

(2) Turn the rudder trim wheel to the left, visually confirm that the rudder trailing edge moves left.

(3) Set the rudder trim wheel in the neutral position.

(4) Turn the rudder trim wheel to the right, visually confirm that the rudder trailing edge moves right.

(5) upon completion of rudder trim rigging and adjustment, when the rudder trim wheel is trimmed left; and, that the rudder trailing edge moves right when the rudder trim wheel is trimmed right.

(6) Make a logbook entry documenting completion of this inspection.

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ELEVATOR AND TAB

1. Troubleshooting

See “Chart 1”.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING ELEVATOR CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between control wheel and elevator.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys.	Install cables correctly.
Resistance to elevator control movement.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Binding control column.	Adjust and lubricate.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Incorrect elevator travel.	Elevator stops incorrectly adjusted.	Adjust stop screws.
Correct elevator travel cannot be obtained by adjusting stops.	Elevator cables incorrectly rigged.	Re-rig elevator cables.
Lost motion between trim control wheel and trim tab.	Cable tension too low.	Adjust cable tension.
	Cables not in place on pulleys.	Install cables properly.
	Broken pulley.	Replace pulley.
	Linkage loose or worn.	Check linkage and tighten or replace.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING ELEVATOR CONTROL SYSTEM

Trouble	Cause	Remedy
Trim control wheel moves with excessive resistance.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys. Provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables properly.
	Trim tab hinge binding.	Lubricate hinge. If necessary, replace.
	Cables crossed or routed incorrectly.	Check routing of control cable.
Trim tab fails to reach full travel.	System incorrectly rigged.	Check and/or adjust rigging.
Trim indicator fails to indicate correct trim position.	Trim indicator unit not adjusted properly.	Adjust trim indicator.

2. Elevator Torque Tube Bearing Changes

   A. Remove nut and washers from end of elevator torque tube assembly.

   B. Remove nuts, bolts and washers that secure the bearing block to the tube support bracket.

   C. Remove bearing.

   D. Replace bearing using the shrink fit process. Bond the bearing with Loctite 290.

   E. Secure bearing block to tube support bracket with nuts, bolts and washers.

   F. Place two washers over the threaded end of the elevator torque tube and secure with nut.

3. Elevator Controls

   A. Removal

      (1) To remove elevator torque tube quadrant:
         (a) Remove elevator downspring.
         (b) Remove the two quadrant cable guards.
         (c) Loosen cables.
         (d) Remove the screw which retains the elevator cable lockplates.
         (e) Remove the cable end retainer (i.e. - lockplate) screws and remove the cables.
         (f) Remove the bolts, nuts and washers that secure the quadrant to the elevator torque tube.
         (g) Remove the quadrant.

(2) To remove the elevator forward cable assemblies:

(a) Disconnect the elevator cable turnbuckles (located forward of the aft elevator quadrant).

(b) Remove the screw that secures the elevator cable end retainers (i.e. - lockplates) to the forward quadrants (on the elevator torque tube).

(c) Pull the cables aft through the fuselage after removing the cable guards at the elevator pulleys and the fin type control cable air seal clips.

(3) To remove the aft elevator cable assemblies:

(a) Disconnect the cable turnbuckles (located forward of the aft elevator quadrant).

(b) Cut safety wire on cable end retainer (i.e. - lockplate) screws. Remove screws.

(c) Remove cable guards and remove cables.

(4) To remove the elevator aft quadrant:

(a) Remove the elevator aft cable assemblies from the quadrant (refer to step 3).

(b) Remove the bolt, nut, washers and bushings which secure the quadrant assembly to the bulkhead (note position of bushings as removed).

(c) Remove the quadrant assembly.

B. Installation

(1) To install the elevator aft quadrant:

(a) Position the elevator quadrant in place.

(b) Refer to “Figure 1” on page 27305 for correct placement of hardware on bolt when installing rear quadrant.

(c) Attach cables to quadrant assembly and secure the cable end retainers (i.e. - lockplate) with screws (see note below). Safety the cable end retainer screws with MS20995-C32 safety wire. Ensure the safety wire is installed so that the wire is inside and clear of the elevator pushrod fork through the full travel of the quadrant.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

(d) Adjust cable guards to be 0.01 ± 0.01 inch from the edge of the quadrant.

(e) Refer to steps (5) and (6).

(2) To install the elevator torque tube quadrant:

(a) Position the quadrant assembly on the torque tube.

(b) Secure quadrant with bolt, nut and washer (two places).

(c) Attach cables to quadrant assembly and secure the cable end retainers (i.e. - lockplate) with screws (see note below). Safety the cable end retainer screws with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket headscrews upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

(d) Adjust cable guard to provide 0.08 ± 0.01 inch of clearance from edge of the quadrant.

(e) Attach the elevator downspring.

(f) Refer to steps (5) and (6).

(3) To install the elevator forward cable assemblies:

(a) Insert the cables through the fin-type air seals on the aft pressure bulkhead and draw them forward through the fuselage, positioning them on the proper pulleys.

(b) Attach cables to forward quadrant and secure the cable end retainers (i.e. - lockplate) with screws (see note below). Safety the cable end retainer screws with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

(c) Adjust cable guards so that there is 0.08 ± 0.01 inch of clearance from edge of the quadrant.

(d) Attach forward cable assemblies to aft cable assemblies at the turnbuckles located just forward of the aft quadrant.

(e) Ensure that all cable guards are in place.

(f) Attach the elevator downspring.

(g) Refer to steps (5) and (6).

(4) To install the elevator aft cable assemblies:

(a) Attach cables to quadrant and secure the cable end retainers (i.e. - lockplate) with screws (see note below). Safety the cable end retainer screws with MS20995-C32 safety wire.

NOTE: As of early 2006, the original MS35265-42 fillister head screw has been replaced by a higher strength MS24674-8 (P/N 520-036) socket head screw. In earlier airplanes, anytime the cable end retainer screws are removed, if the original fillister head screws are still installed, they must be replaced with the socket head screws upon reassembly. Additionally, when using the MS24674-8 (P/N 520-036) socket head screws, torque to 12 to 15 in.-lbs.

(b) Attach the rear cables to the forward cables at the turnbuckles.

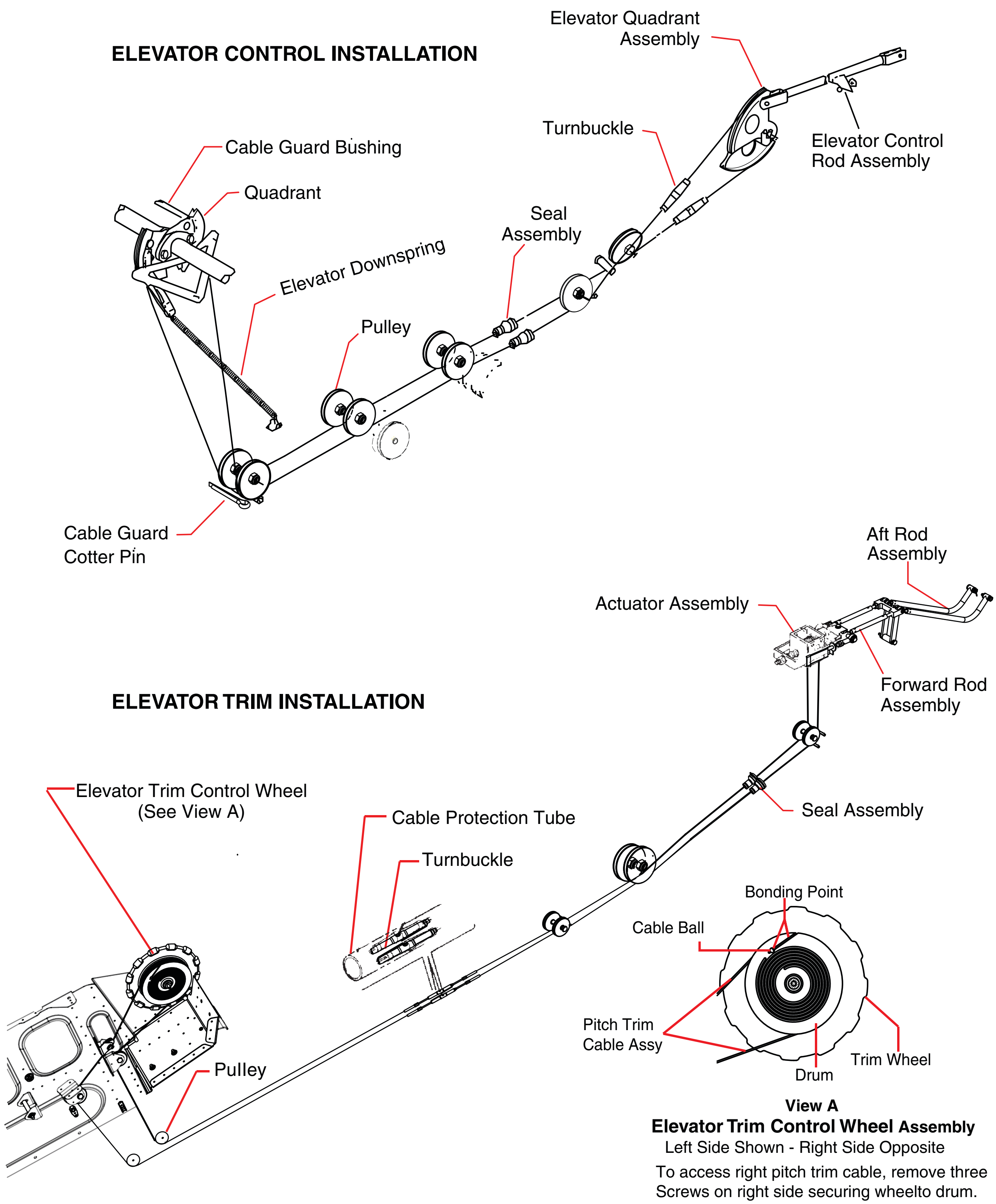
(c) Reinstall cable guards and adjust them to provide 0.01 ± 0 .01 inch of clearance from the edge of the quadrant.

(d) Refer to steps (5) and (6).

(e) Attach the elevator downspring (refer to “Figure 1”).

(5) Set cable tension.

(6) Check control system for proper rigging and operation.



Elevator and Elevator Trim Installation
Figure 1

C. Rigging and Adjustment

NOTE: The neutral position of the elevator is when the leading edges of the aerodynamic balances are lined up with the leading edges of the horizontal stabilizer.

(1) Adjust the elevator stops to obtain the travel specified in "Chart 2" on page 27004.
(2) Lock the elevator in its neutral position using a suitable tool.
(3) Position the pilot's control wheel with the forward face of the wheel mounting boss as shown in Figure 3, 27-10-00 and secure with a suitable tool.
(4) Adjust the cable turnbuckles evenly to obtain cable tension specified in "Chart 2" on page 27004.
(5) Remove the locking devices from the control wheel and elevator.
(6) With the elevator held against the trailing edge down primary (aft) stop, push the pilot's wheel full forward. The control wheel travel between the primary (aft) and secondary (forward) stops should be 0.30 ± 0.10 inch. There is no adjustment on the secondary stop.

NOTE: If necessary, use the cable rigging turnbuckles to bring the control wheels into their neutral position.

(7) With the elevator physically held against the trailing edge up primary (aft) stop, adjust the secondary stop on the channel ahead of the instrument panel to provide a cushion (gap) of 0.06–0.09 inch. Lock the jamnut.

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF ELEVATOR. UPON COMPLETION OF ELEVATOR RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE ELEVATOR TRAILING EDGE MOVES UP WHEN THE WHEEL IS PULLED BACK; AND, THAT THE ELEVATOR TRAILING EDGE MOVES DOWN WHEN THE WHEEL IS PUSHED FORWARD.

(8) Proceed to "Post-Rigging Inspection", below.

D. Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any elevator control cable or pushrod replacement, disconnect, or rigging perform the following post-rigging inspection:

(1) Pull the pilot's control wheel FULL AFT, as if to nose the airplane UP, until reaching the primary stop. Visually confirm the elevator is deflected UP. The trailing edge of the elevator will be noticeably and significantly ABOVE the horizontal stabilizer.
(2) Push the pilot's control wheel FULL FORWARD, as if to nose the airplane DOWN, until reaching the primary stop. Visually confirm the elevator is deflected DOWN. The trailing edge of the elevator will be noticeably and significantly BELOW the horizontal stabilizer.
(3) Repeat the above steps for the copilot controls.
(4) Make a logbook entry documenting completion of this inspection.

4. Elevator Trim Controls

A. Trim Wheel and Cables

(1) Removal

To remove elevator trim controls in the console:

(a) Remove access panel on left side of console.

(b) Disconnect elevator trim cables by:

1) Remove the center access panels on the floor to gain access to the plastic tube that protects the elevator trim cables.

2) Remove the pins from both ends of the plastic tube.

3) Slide the tube forward.

4) Rotate the elevator trim control wheel until one of the trim cable turnbuckles clears the aft end of the tube.

5) Tie a pull rope forward of the exposed turnbuckle on the forward cable.

NOTE: Tension must be applied to the ends of the trim cables as soon as the turnbuckles are disconnected to prevent the cables from unwrapping from the trim wheel drum or the trim actuator, and to prevent the cables from fouling at any of the pulleys.

6) Disconnect the cables, keeping tension on both ends of cable.

7) Have an assistant in the cockpit rotate the elevator trim control wheel until the opposite elevator trim turnbuckle clears the aft end of the plastic tube. Keep tension on both ends of the disconnected cable while the assistant is rotating the trim wheel.

8) Have the assistant tie another pull rope forward of the turnbuckle on the remaining cable.

9) Have the assistant disconnect his cables, keeping tension on both ends of the cable.

10) Transfer the cable ends to one man. Have him keep tension on all four cable ends.

11) Have the remaining man slide the plastic tube aft until the cables are exposed aft of the F.S. 124.125 frame.

(c) Remove the nut and washer from trim wheel pivot bolt.

(d) Push the trim wheel pivot bolt out the right hand side of the quadrant.

(e) Remove the trim wheel from the console, pulling the control cable along with it.

(f) To disassemble the elevator trim wheel:

1) Remove the three screws which attach the cable drum to the wheel.

2) Remove the cable from the drum.

(2) Installation

(a) To assemble the elevator trim wheel:

1) Place the cable ends in the drum and bond using DP-105 (P/N 279-131).

NOTE: DP-105 is a Type IV adhesive and is supplied in a Duo-Pak cartridge for the Scotch-Weld EPX applicator system, the correct proportions of resin and curing agent are mixed directly by the EPX static mixer tip.

2) Bonding procedure:

a) Clean

1] Metal parts to be bonded shall be cleaned and primed.

2] Final cleaning of the primed surface prior to adhesive application shall be accomplished using MEK, isopropyl alcohol or acetone.

b) Application

1] Coat each surface to be mated with 3M Scotch-Weld DP-105 (P/N 279-131) clear epoxy adhesive.

2] Press parts together immediately (within 3 minutes) and hold a few seconds until “grab” begins.

3] Remove excess cement with a clean wipe.

4] Allow approximately 20 minutes (approximately 10 minutes after adhesive is tack free) for sufficient handling strength to remove pressure. Maximum strength is achieved in 24-48 hours.

c) Curing

5] Full cure takes 48 hours minimum at (73°F) room temperature.

6] After 20 minutes at room temperature the bonded assemblies have enough handling strength to remove pressure with careful handling and moving.

3) Secure the cable drum to the wheel, using the three screws.

4) Wrap the cable ends around the drum 3 1/4 turns in opposite directions.

(b) Position the trim wheel/cable in place in the console.

(c) Slide the trim wheel pivot bolt through the pivot holes in the trim wheel and the rudder trim indicator bracket.

(d) Secure the pivot bolt with the nut and washer.

(e) Connect the elevator trim cables:

1) Have the assistant keep tension on the trim cables while the cable blocks at F.S. 207.00 and F.S. 124.125 are removed.

2) Slide the plastic tube forward.

3) Connect one set of cables at the turnbuckles and remove the pull rope. Keep tension on the remaining cables.

4) Have the assistant rotate the elevator trim control wheel until the remaining (unconnected) turnbuckle clears the aft end of the tube.

5) Connect the remaining cables and remove the pull rope. Set cable tension to 12 ± 2 pounds.

NOTE: Both the up and down cable turnbuckle assemblies should be within 0.10 of an inch of the same length upon completion of adjustment.

6) Slide the tube in place and insert the pins in both ends of the tube.

7) Check to ensure that the cables have not loosened on the trim drum or actuator. Check that they have not fouled on any of the pulleys.

8) Check the system for restrictions or binding.

9) Reinstall the floor access panels that were previously removed.

(f) Ensure that the pulley cable guards are installed at all elevator trim cable pulleys.

(g) Replace the access panel on the console.

(h) Check operation of elevator trim controls. See Caution under Rigging.

B. Elevator Trim Actuator

(1) Removal

(a) Block the cables at the actuator drum.

(b) Disconnect the trim cable turnbuckles (inside the plastic tube under the floor). (See Elevator Trim Controls, Removal, above.)

(c) Cut safety wire and remove bolts that attach actuator to elevator trim forward rod assembly.

(d) Remove four bolts and washers that secure the actuator to its supporting bracket.

(e) Remove actuator and trim cable from aircraft.

(2) Installation

(a) Position actuator/trim cable assembly on actuator bracket. Fasten in place with bolts and washer.

(b) Attach elevator trim forward rod assembly to actuator with bolts and washers. Safety bolts.

(c) Connect the trim cables. (See Elevator Trim Controls, “Installation” on page 27307.)

(d) Ensure that cables are in pulleys and that pulley cable guards are installed.

(e) Attach three cable clips (per seal) to the fin type control cable air seals in the aft pressure bulkhead.

(f) Set cable tension.

(g) Check system for smooth and proper operation.

(3) Disassembly

(a) Remove nut and washer from the end of the actuator screw.

(b) Remove the bolt, nut and washer that secures the actuator screw to the actuator frame.

(c) Turn the screw out of the housing assembly and then out of the frame.

(d) Remove the drum from the housing assembly by removing the bolts which secure the aft retainer to the housing.

(e) Unwrap cable from drum (if desired).

(4) Assembly

(a) Place trim cable in slot in trim drum. Cable end Y should be 17.5 inches longer than cable end X.

(b) Wrap each end of the cable nine and one-quarter turns toward the center of the drum. One series of wraps should be clockwise and the other series should be counterclockwise. Hold the cable in place.

(c) Insert the drum into the housing assembly.

(d) Secure the aft retainer and guide to the aft side of the housing with the screws, nuts and washers.

NOTE: Refer to “Figure 2” for information on setting trim drum end play.

(e) Place the housing/drum assembly into the actuator frame.

(f) Insert the actuator screw through the frame and then into the housing/drum assembly.

(g) Rotate the screw until the housing/drum assembly is positioned as shown in “Figure 2” on page 273010. Cable end “Y” should be 17.5 inches longer than cable end “X” after installation and wrapping on the drum before fastening the screw in position.

(h) Attach the aft end of the actuator screw to the actuator frame by installing the bolt, nut and washer. The bolt shank should point in the same direction as the cable ends.

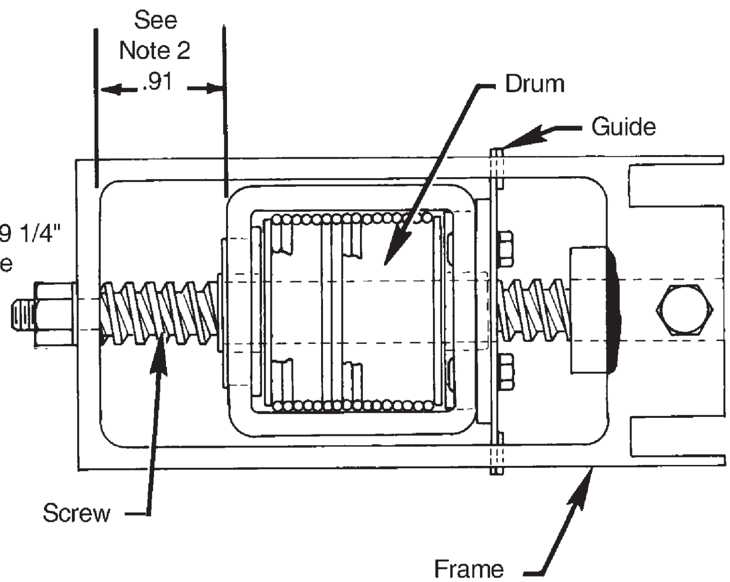
(i) Attach the forward end of the actuator screw to the frame with the nut and washer.

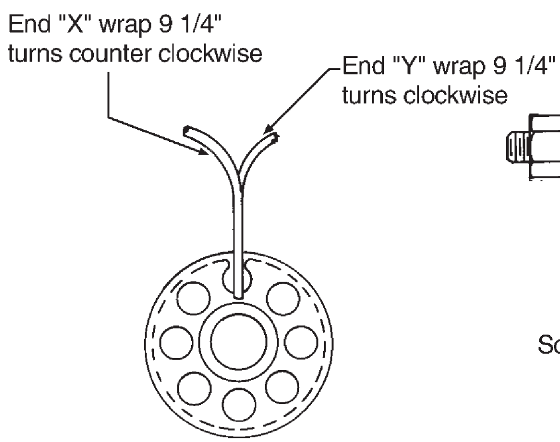
(j) Check for free, smooth full range travel.

NOTES:

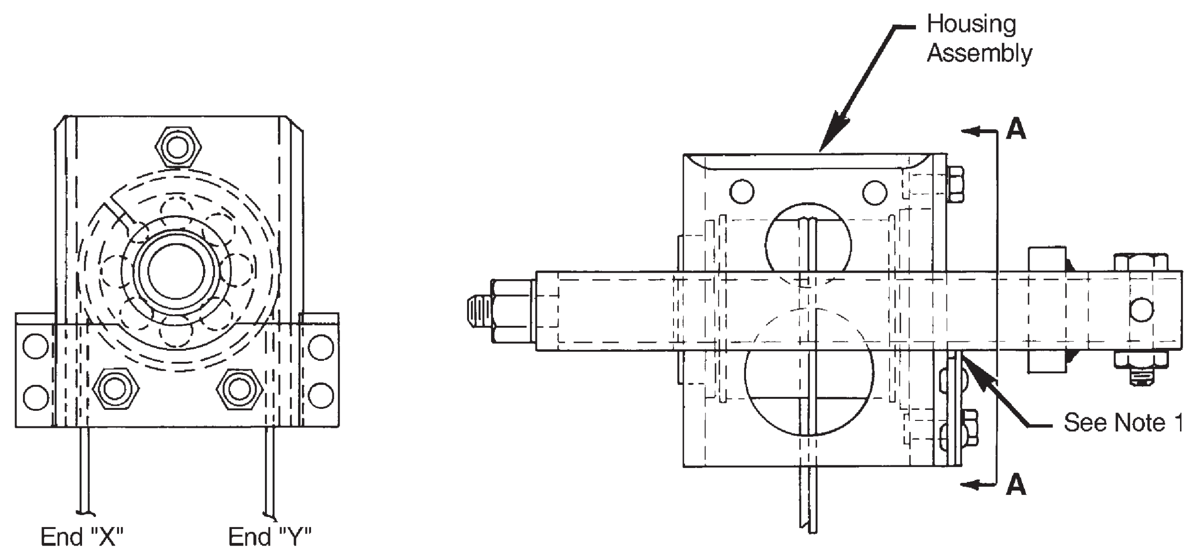
82945 R
82799 N

1. With the frame contacting the guide on one side, check for .025/.005 clearance.
2. Cable end “Y” should be 17.5 inches longer than cable end “X” prior to fastening the actuator screw in position.
3. Add 59486-2, -3, or -4 washers, as required, to reduce drum end play to .001 to .003 inches.
4. Add 82916-2 shims, as required, to increase drum end play to .001 to .003 inches.





Drum
(View looking forward)



View A - A

Elevator Trim Actuator Assembly
Figure 2

C. Elevator Trim Tab Push Rod Corrosion Control

Inspect elevator trim rods and reapply Dinatrol AV 8 each five years. See 55-20-00, Elevator Trim Tab Push Rod Corrosion Control.

D. Rigging and Adjustment

NOTE: The neutral position of the elevator tab is with the tab trailing edge lined up with the elevator trailing edge.

(1) Check the tension of the elevator trim tab cables. Adjust the tension as specified in "Chart 2" on page 27004. The up and down cable turnbuckle assemblies should be within 0.10 inch of the same length upon completion of adjustment.

NOTE: Obtain access to the turnbuckles by rolling the trim wheel to the "full nose up" position and sliding the tube that surrounds the cables fore and aft, as required.

(2) Place the elevator tab in the neutral position and set the tab indicator at the neutral position using a suitable locating fixture.

(3) Move the elevator trim wheel aft (nose up) until the positive stop (spacer) on the actuator frame engages the trim drum housing. Check to ensure that the tab indicator guide pin has not bottomed out in its slot in the trim wheel.

CAUTION: DO NOT ADJUST THE TWO AFT CURVED RODS WHICH ATTACH TO THE TAB. THESE RODS MUST NOT BE CHANGED FROM THEIR ORIGINAL LENGTH OF 9.12 INCHES BETWEEN THE CENTERS OF THE ROD END HOLE AND THE MOUNTING BUSHING HOLE. A CHANGE IN THEIR LENGTH WOULD ALTER THE PROPER TAB OPERATING GEOMETRY.

(4) Set the tab to the Elevator Tab (Elevator Neutral) Trailing Edge Down angle specified in "Chart 2" on page 27004, by adjusting the two straight rods which penetrate the aft fuselage bulkhead. The rods must be within 0.010 inch of the same length upon completion of adjustment. Lock the jam nuts on the rods.

(5) Move the elevator trim wheel forward (nose down) until the elevator tab is at the Elevator Tab (Elevator Neutral) Trailing Edge Up angle specified in "Chart 2" on page 27004. Check to ensure that the tab indicator guide pin has not bottomed out in its slot in the trim wheel. Set the adjustable stop mounted on the stabilizer spar to maintain the specified tab angle. Lock the jam nut.

(6) Operate the elevator through its full travel range with the tab set at both full nose up and full nose down positions. Operation should be smooth and free.

(7) With the tab neutral and the elevator held securely against each stop, determine the free play of the tab. Total free play measured at the tab trailing edge should not exceed 0.06 inch.

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF ELEVATOR TAB. UPON COMPLETION OF ELEVATOR TRIM RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE ELEVATOR TAB TRAILING EDGE MOVES UP WHEN THE TRIM WHEEL IS TRIMMED DOWN; AND, THAT THE ELEVATOR TAB TRAILING EDGE MOVES DOWN WHEN THE TRIM WHEEL IS TRIMMED UP.

(8) Proceed to "Post-Rigging Inspection" on page 273012.

E. Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of elevator trim system component replacement, disconnect, or rigging and adjustment perform the following post-rigging inspection:

(1) Move the elevator trim wheel forward (nose down), visually confirm that the elevator tab trailing edge moves up.

(2) Return the elevator trim wheel to neutral.

(3) Move the elevator trim wheel aft (nose up), visually confirm that the elevator tab trailing edge moves down.

(4) Make a logbook entry documenting completion of this inspection.

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5. Stall Warning System

A. Description and Operation

WARNING: PHYSICAL MANIPULATION OF LIFT TRANSDUCER MAY RESULT IN ERRONEOUS STALL WARNINGS NECESSITATING CALIBRATION OF SYSTEM.

The stall warning system's major components consist of a lift transducer located in the leading edge of the left wing (See 39-20-00 Figure 21) and a lift computer located on the aft electronics shelf (See 39-20-00 Figures 23, 24 and 25). The sensing source of the transducer protrudes into the air stream and during flight is positioned by local airflow velocity and direction. By correlating lift with airflow characteristics at the stagnation point on the wing, the lift transducer measures changes in angle-of-attack, or airspeed deviations from an airspeed represented by a reference angle-of-attack.

The output signal, combining angle-of-attack and local dynamic pressure, enters the lift computer which contains power regulators, signal processor and control circuitry. Provisions are included for monitoring the entire stall warning system.

B. Lift Transducer Assembly

(1) Removal

(a) Remove the four screws which hold lift transducer to leading edge of wing.

(b) Pull lift transducer partially out of opening in wing and disconnect wires at connector.

NOTE: A special tool (J-Block Extract Tool) is required to remove the connector without damage. See “Figure 6” on page 273021 to fabricate.

(c) Remove lift transducer.

(2) Installation

(a) Connect wires from transducer to harness.

(b) Position lift transducer in the opening in the wing and secure with the four screws.

(c) Check lift transducer for operation per Stall Warning Functional Test.

C. Dual Warning Unit (Horn)

(1) Removal

(a) Disconnect wires coming from warning unit.

(b) Remove the four screws and washers that secure the unit to the bulkhead.

(2) Installation

(a) Position dual warning unit on the bulkhead and secure with screws/washers.

(b) Connect electrical leads to unit (refer to electrical schematic in 91-27-30).

D. Stall Warning Functional Test

(PIR-PPS60089-2, Rev. H)

(1) Locate the stall warning test switch on the pilot's instrument panel and the lift transducer which is located in the inboard left wing leading edge.

(a) Activate the test switch.

CAUTION: SURFACE OF PLATE IS HOT WHEN ACTIVATED AND CAN BURN, DO NOT TOUCH FOR A PROLONGED TIME.

(b) Verify that the transducer vane moves forward (up); the stall warning activates.

NOTE: For the Heat to be felt from the plate one of the OAT probes must be less than 5°C / 41°F.

(c) Make ice bath in a small container. Submerge the tip of either OAT probe in the ice bath. Activate the stall warning heat switch on the instrument panel. Check that lift transducer vane and face plate become warm.

(2) Locate the stall warn heat switch on the instrument panel (early models) or overhead panel (later models).

CAUTION: DO NOT ACTIVATE STALL WARN HEAT SWITCH FOR MORE THAN FIVE (5) SECONDS.

(a) To check high heat to lift transducer vane and mounting plate:

1) For S/N's 4636446 and up, S/N's 4692055 and up and earlier airplanes with Piper Kit P/N 88452-002 installed, remove weight on left main gear squat switch.

2) For S/N's 4636001 thru 4636445 and S/N's 4692001 thru 4692054 without Piper Kit P/N 88452-002 installed, depress left main gear uplock switch.

(b) Activate the stall warn heat switch.

(c) Check that the lift transducer vane and mounting plate become warm.

E. Flap Switch Input Test

(1) Set flap to 0° with aircraft power on.

(2) Remove electrical connector P376 from lift computer.

(3) With continuity tester monitor pins 4 and 5 of P376. Verify open circuit.

(4) Move flap position switch to 10°. Verify continuity between pins 4 and 5.

(5) Move flap position switch to 20°. Verify continuity between pins 4 and 5.

(6) Move flap position switch to 36°. Verify open circuit between pins 4 and 5. Verify continuity between pins 4 and 6.

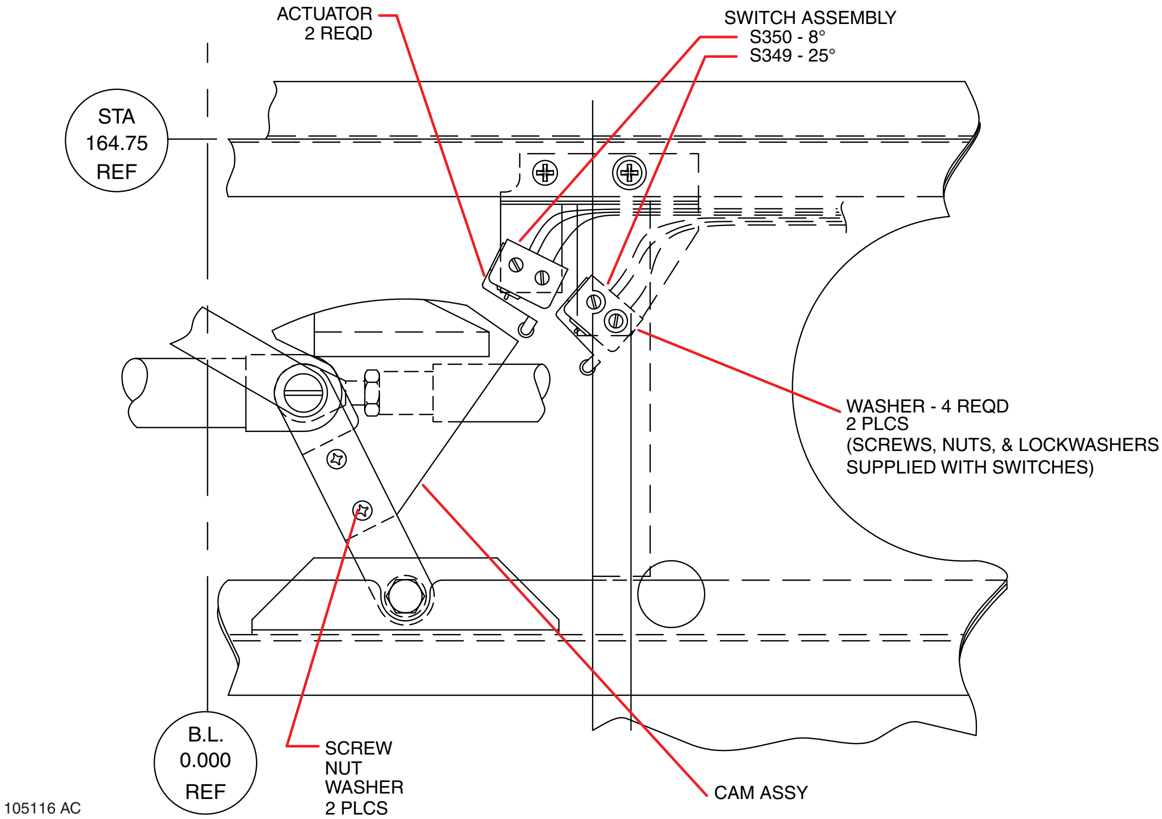
(7) Move flap position switch back to 20°. Verify open circuit between pins 4 and 6. Verify continuity between pins 4 and 5.

(8) Move flap switch to 0°. Verify open circuit between pins 4 and 5.

(9) Advance throttle lever to full throttle. Verify continuity between pins 3 & 4.

(10) Retard throttle lever to idle. Verify open circuit between pins 3 & 4.

(11) If all tests pass reinstall P376 on lift computer. If all do not pass verify flap microswitch installation and adjustment (refer to "Figure 3" on page 273016). Repeat steps 3 thru 10.



Stall Warn Flap Compensation Switches
Figure 3

F. Force Applicator Kit Test

The lift computer is factory set by Safe Flight Instrument Corp. to produce the required outputs to activate the stall horn. Final adjustments are then made to each installation by Piper during fight test. At the completion of individual system adjustment by Piper the actual tip gram value will be recorded in the aircraft logbook as follows:

STALL WARNING (Tipgrams ± 1/2)

		TIP GRAMS	
POWER	**FLAPS**	**ACTUAL SETTINGS**	**EXAMPLE**
Off	Full Up 0°	x.x	(+1.5)
Off	10°-20°	x.x	(+1.0)
Off	Full Down 36°	x.x	(0)
On	Full Up 0°	x.x	(+0.5)
On	10°-20°	x.x	(0)
On	Full Down 36°	x.x	(-0.75)

Should it become necessary to check the values shown in the airplane log book, use Safe Flight Instrument Corp's. Force Gram Applicators (FGA), P/N 1952-1 or P/N 1952-3, measure the tip grams required to sound the stall warning audible alert. Use the FGA with the lowest range appropriate for the desired applied tip grams (see "Figure 5" on page 273018), to measure the "tip-grams" of force settings required at the lift transducer vane to energize the signal.

(1) Attach the FGA to the lift transducer cavity as shown in Safe Flight's Force Applicator Kit operation manual (see "Figure 5" on page 273018). The squat switch shall be in the in-flight condition.

(a) Position the cockpit pedestal power lever to the power off position and flaps to the full up position. The prestall horn shall activate at tip grams recorded in the airplane log book.

(b) Place the cockpit pedestal power lever in the power off position and flaps to the 10°-20° position. The prestall horn shall activate at tip grams recorded in the airplane log book.

(c) Place the cockpit pedestal power lever in the power off position and the flaps at the full (36°) down position. The pre-stall horn shall activate at tip grams recorded in the airplane log book.

(2) To make necessary adjustments, remove the access plate from the lift computer. Locate the four adjustment screws number "0, 1, 2, and 3" (refer to "Figure 4" on page 273017).

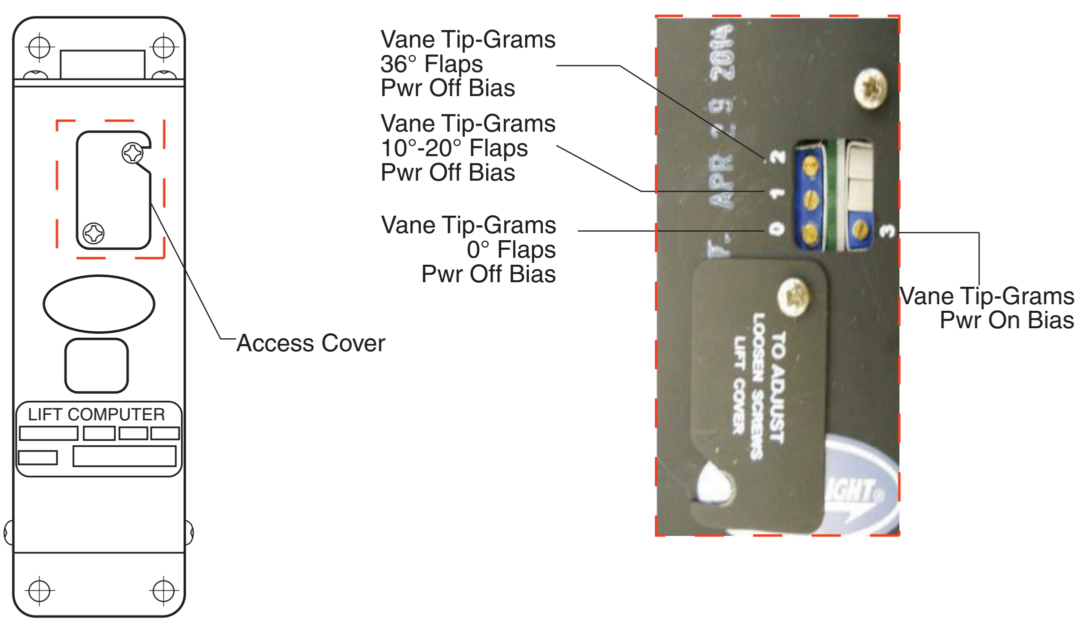
G. Positive Tip Gram Adjustment

(1) If the stall warning sounds with a lesser force than the desired tip grams, the stall warning settings can be re-calibrated by the following instructions.

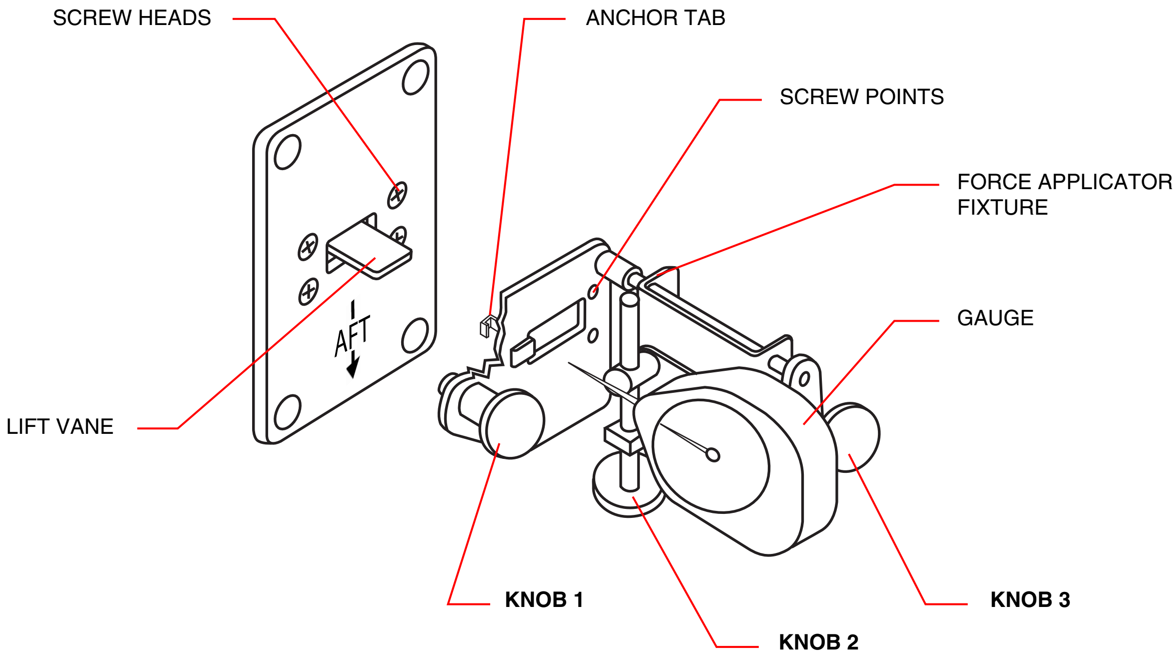
(a) Set the Force Gram Applicator (FGA), P/N 1952-1 or P/N 1952-3 (see "Figure 5" on page 273018) to apply the desired tip grams to the lift transducer vane. Slowly, turn the appropriate pot Counterclockwise (CCW) until the stall warning stops sounding. While turning the pot CCW, every ¼ of a turn, apply a downward force, with your finger, to the lift transducer vane to stop the stall warning sounding. Slowly, release the force on the lift transducer vane until it contacts the FGA. If stall warning is not sounding with the desired tip grams applied to the lift transducer vane, slowly, turn the pot Clockwise (CW) until the stall warning sounds.

(b) Reset the FGA to apply +2 tip grams of force greater than the desired tip grams. Slowly, reduce the force on the lift transducer vane from the FGA until the stall warning sounds. The force being applied by the FGA on the lift transducer vane should be the desired tip grams ±0.5 TG.

(c) Repeat the previous step to verify proper stall warning setting before proceeding. If stall warning setting is not correct, repeat steps a and b.



Lift Computer "Tip Grams" Adjustment
Figure 4



FGA	RANGE
1952-1	0.3 – 3.0 TG
1952-3	2.0 – 15.0 TG

KNOB 1 Tighten to secure to transducer.

KNOB 2 Apply force to vane:

Negative TG (-) Up or Forward

Positive TG (+) Down or Aft

KNOB 3 Used to position gauge In or Out.

Force Gram Applicator (FGA)
Figure 5

(2) If the stall warning does not sound after reaching the desired tip grams plus +0.5 TG, the stall warning can be re-calibrated by the following instructions.

(a) Set the FGA to apply the desired tip grams to the lift transducer vane. Slowly, turn the appropriate pot CW until the stall warning sounds.

(b) Reset the FGA to apply +2 tip grams of force greater than the desired tip grams. Slowly, reduce the force on the lift transducer vane from the FGA until the stall warning sounds. The force being applied by the FGA on the lift transducer vane should be the desired tip grams ±0.5 TG.

(c) Repeat the previous step to verify proper stall warning setting before proceeding. If stall warning setting is not correct, repeat steps a and b.

H. Negative Tip Gram Adjustment

(1) If the stall warning sounds before reaching a lesser force than the desired tip grams, the stall warning can be re-calibrated by the following instructions.

(a) Set the FGA (see "Figure 5" on page 273018) to apply the desired tip grams to the lift transducer vane. Slowly, turn the appropriate pot CCW until the stall warning stops sounding. While turning the pot CCW, every ¼ of a turn, reduce the force the FGA is applying to the lift transducer vane to 0 TG and retract the FGA. Apply a downward force, with your finger, to the lift transducer vane to stop the stall warning sounding. Slowly, release the force on the lift transducer vane until it is neutral. Re-engage the FGA and re-apply the desired tip grams to the lift transducer vane. If the stall warning is not sounding when you reach the desired tip grams, slowly, turn the appropriate pot CW until the stall warning sounds.

(b) Reduce the force the FGA is applying to the lift transducer vane to 0 TG and retract the FGA. Apply a downward force, with your finger, to the lift transducer vane to stop the stall warning sounding. Slowly, release the force on the lift transducer vane until it is neutral.

(c) Engage the FGA and slowly, re-apply a negative force on the lift transducer vane until the stall warning sounds. The force being applied by the FGA on the lift transducer vane should be the desired tip grams ±0.5 TG.

(d) Repeat the previous step to verify proper stall warning setting before proceeding. If stall warning setting is not correct, repeat steps a, b and c.

(2) If the stall warning does not sound after the desired tip grams the stall warning can be re-calibrated by the following instructions.

(a) Slowly, turn the appropriate pot CW until stall warning sounds.

(b) Reduce the force the FGA is applying to the lift transducer vane to 0 TG and retract the FGA. Apply a downward force, with your finger, to the lift transducer vane to stop the stall warning sounding. Slowly, release the force on the lift transducer vane until it is neutral.

(c) Engage the FGA and slowly, re-apply a negative force on the lift transducer vane until the stall warning sounds. The force being applied by the FGA on the lift transducer vane should be the desired tip grams ±0.5 TG.

(d) Repeat the previous step to verify proper stall warning setting before proceeding. If stall warning setting is not correct, repeat steps a, b and c.

After the stall warning functional test, flap switch input test and stall warning calibration is done, re-install the access plate from the lift computer. Return squat switch to on ground condition.

I. Lift Computer (LC) Test

(1) If available, connect the Breakout Box (Safe Flight P/N SK100360) between the lift computer and the aircraft wiring. Locate the breakout box so that the test jacks (which connect to the like numbered pin) are accessible. If no breakout box is available, the pins may be accessed directly. See “TABLE 1”.

(2) Attach the Force Applicator to the lift transducer as shown in “Figure 3” on page 273016 so as to apply forces to the vane as specified in the following test sections. Use the gauge with the lowest range adequate for the readings. Forward forces on the vane are designated as minus (-): aft forces are designated as positive (+).

(3) Set the aircraft flaps to the UP position for 0° readings, MID position for 10° and 20° readings and DOWN for 36° readings.

(4) Bypass the OLEO (i.e. - squat) switch function to the IN AIR condition. A jumper between PINS 2 (GND) and 7 (OLEO) will accomplish this.

(5) Apply 28V DC ± 20% DC power to the system by actuating the proper circuit breakers. Do not apply power to the lift transducer heaters while making these tests.

TABLE 1
LIFT COMPUTER PIN OUTS

Pin	Function	Voltage Reading
1	+28 VDC Power Source	Constant
2	DC Power Ground	
3		
4	Flaps Excitation	28V Constant
5	10 and 20 Flaps	28V only at these settings
6	36 Flaps	28V only at this setting
7	OLEO - Connected to ground when IN AIR	
8	SELF TEST - Momentary ground for test	
9	VALID FLAG	+28 VDC if system is valid
10	ESSENTIAL BUS	+28 VDC Constant (3 Amp Breaker)
11		
12	Stall Warning Voltage	+28 VDC to warning device
13 - 15		
16	Lift Transducer (LT) Self Test	+15 VDC for ON GND Self Test to LT
17	Fail Warning	+28 VDC if system is not valid
18 - 21		
22	TEST DATA	Varying voltage controlled by lift transducer
23	SIGNAL GROUND	
24	LIFT TRANSDUCER EXCITATION	10V RMS 700 Hz power to LT
25	LIFT TRANSDUCER EXCITATION	Pin 25 is high
26	FAST	Lift Transducer Signals to Computer
27	COMMON	Lift Transducer Signals to Computer
28	SLOW	Lift Transducer Signals to Computer

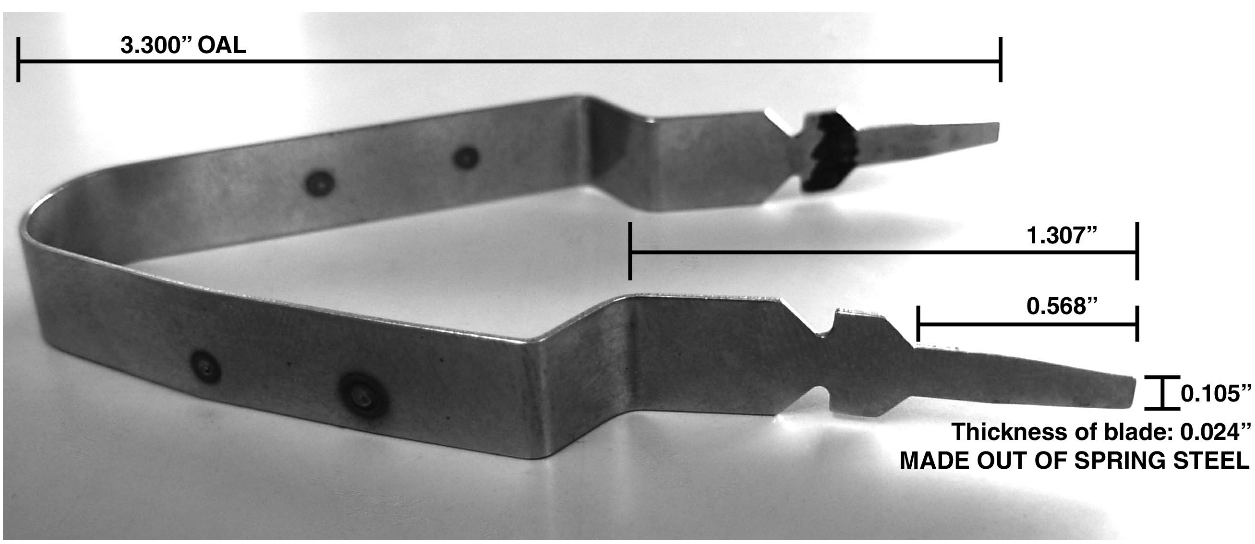
J. Lift Transducer (LT) Test

(1) Check that the AC voltage measured between PIN-24 and PIN-25 (PIN-25 is High) is 10.0 ± 0.5 V.

(2) Check the lift transducer and the lift computer phase detector by comparing the voltages of "TABLE 2" below, PIN-22 and PIN-23 (PIN-23 = Low). The voltage across pins PIN-26 and PIN-28 (PIN-26 = Low) is an AC voltage varying in polarity and magnitude with the LT tip gram force.

TABLE 2
LIFT TRANSDUCER TEST SPECIFICATIONS

Tip Grams	PIN - 26 (LOW) PIN - 28 (HI) AC Mili-Volts		PIN - 22 (HI) PIN - 23 (LOW)	
	Ref.	Tolerance	DC Volts	Tolerance
-6	427	±20	9.20	±0.75
0	279	±20	6.00	±0.75
12	0	±20	0.00	±0.75
24	279	±20	-6.00	±0.75



J-Block Extract Tool
Figure 6

K. Stall Warning Flight Test Procedure (PIR-FTP2001-8, Rev. F., FTP2001-8-1, Rev. E. FTP2007-1, Rev. C., FTP2009-1, Rev.J., and FTP2009-2, Rev. G.)

NOTE: Required whenever a "replacement" wing is installed.

This test is performed with gear down, flaps set as indicated, and power ON = 31 In. Hg. @ 2400 RPM. Fly airplane and at a rate of 1 KT/SEC maximum record the following warning and stall speeds:

Flaps	Power	Trim Speed	Warning Limits	Record Warning / Stall
0°	IDLE	110	5 - 10	________/________
0°	ON	110	5 - 10	________/________
20°	IDLE	100	5 - 10	________/________
20°	ON	100	5 - 10	________/________
36°	IDLE	90	5 - 10	________/________
36°	ON	90	5 - 10	________/________

(1) Verify that the stall warning system performs within the following limits:

Stall warning begins at 5 to 10 KIAS prior to stall and continues until stall occurs.

(2) If the stall warning audible alert does not sound at the appropriate airspeeds, power settings and flap configurations, the lift computer must be calibrated while in flight.

(a) Remove the access plate from the lift computer. Locate four potentiometer (Pot) screws numbered 0, 1, 2 & 3, (refer to "Figure 4" on page 273017).

1) Pots 0, 1 & 2 adjust the stall warning settings in the power OFF position for 0°, 10°- 20°, 36° flap configurations respectively.

2) Pot 3 adjusts the stall warning settings in the power ON position for all flap configurations.

(a) To decrease the airspeed at which the stall warning sounds, turn the appropriate Pot Counterclockwise (CCW).

(b) To increase the airspeed at which the stall warning sounds, turn the appropriate Pot Clockwise (CW.)

(c) Re-install the access plate from the lift computer after all calibrations are made.

(3) If not able to adjust the stall warning system sufficient to achieve the limits above, recheck the "replacement" wing installation to ensure it is installed and rigged correctly.

After the test flight, if any Pot calibrations were required the actual tip gram values at which the stall warning sounds for each power setting and flap configuration must be confirmed.

L. Post Flight Test Stall Warning Confirmation

Determine if a positive or negative force will be required to make the stall warning sound by the following instructions.

(1) Defeat the squat switch to simulate the inflight condition.

(2) Set the power setting and flap configuration at the desired positions. If the stall warning is sounding it will required a positive force, if stall warning is not sounding it will require a negative force.

For a positive force, attach the FGA to the lift transducer per F and G above. Slowly, apply a positive force to the lift transducer vane until the stall warning stops sounding. Add an additional +2 TG to the lift transducer vane. Slowly, reduce the force on the lift transducer vane from the FGA until the stall warning sounds. The force being applied by the FGA on the lift transducer vane shall be considered the stall warning setting for this power setting and flap configuration. Repeat this procedure to verify stall warning setting before proceeding. Record the tip gram reading in the aircraft logbook.

For a negative force, attach the FGA to the lift transducer per "F" on page 273016 and "H" on page 273019. Slowly, apply a negative force on the lift transducer vane until the stall warning sounds. The force being applied by the FGA on the lift transducer vane shall be considered the stall warning setting for this power setting and flap configuration. Repeat this procedure to verify stall warning setting before proceeding. Record the tip gram reading in the aircraft logbook.

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FLAPS

1. Flap Drive System

   Refer to “Figure 1” on page 27502.

   The flaps are driven by an electric motor-actuator unit. The motor-actuator is mounted below the floor at fuselage station (F.S.) 167.97. Remove the seat, cabinet (if necessary), and floor boards to gain access to the motor.

   The drive system contains two movable micro-switches on a cam mechanism (one each for up and down flap movement), warning horn micro-switch on cam bracket, current sensor, relay, and mechanical linkages. A removable cover on the right forward floor gives access to the switches. The sensor and relay are located on a plate assembly aft of motor. The cover is secured with velcro tabs for easy removal.

   Pushrods are connected to the actuator at both ends. These pushrods extend out through the fuselage and connect to bellcranks in the wings at wing station (W.S.) 29.00. Additional pushrods extend out along the wing rear spar to an idler arm at W.S. 64.00 and on to another bellcrank at W.S. 93.00. Pushrods connected to the bellcranks at W.S. 64.00 and W.S. 93.00 connect the flaps to complete the drive system.

   To ensure a good pressure seal at the fuselage exit points of the flap drive pushrods, a bellows is connected to the pushrods and tube assemblies which are part of the fuselage structure, thus sealing the opening and preventing any air loss from the pressurized cabin.

   The flap selector provides four positions: Full up (0°), 10°, 20°, and full down (36°). All positions are preselected by the flap control lever on the instrument panel. A flap position warning system is provided to warn when the flaps are extended to or beyond the approach position without the landing gear fully extended and locked. Flap position indication is provided as follows:

   A. S/N’s 4636001–4636313, a cable attached to the flap drive system drives a mechanical pointer installed to the left of the flap control lever.

   B. S/N’s 4636314–4636459, 4636461–4636462, an electrical dial gauge is installed adjacent to the flap control lever. The gauge is driven by a linear potentiometer in the flap drive system.

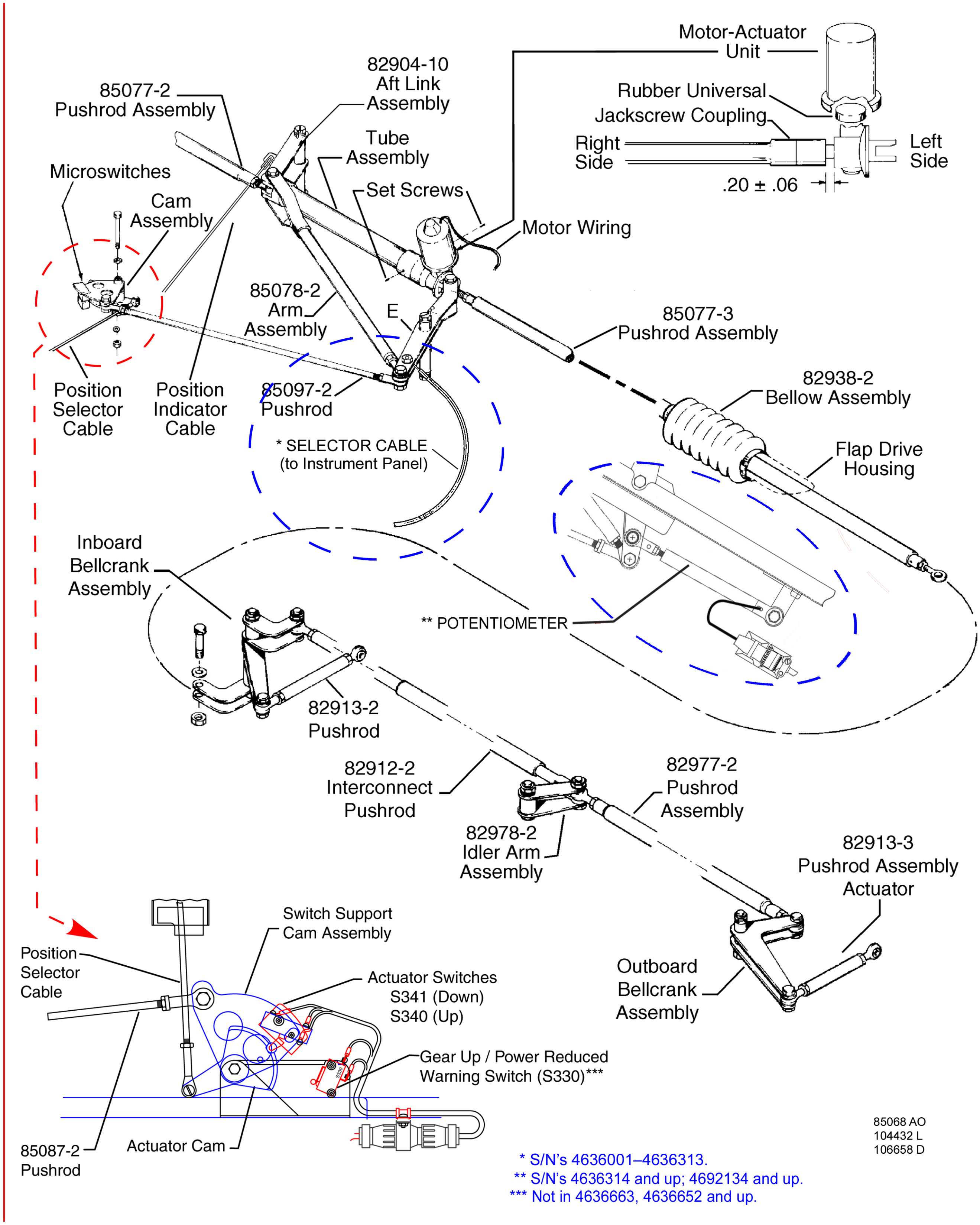
   C. S/N’s 4636460, 4636463 and up, 4692134 and up; flap position is displayed in the MFD, driven by a linear potentiometer in the flap drive system.

2. Troubleshooting

   See “Chart 1”.

CHART 1
TROUBLESHOOTING FLAP CONTROL SYSTEM

Trouble	Cause	Remedy
Flaps fail to extend or retract.	Motor-Actuator inoperative.	No electrical power to motor. Check circuit breaker.
Flaps not synchronized or fail to move evenly when retracted.	Incorrect rigging of system.	Adjust flaps.



Flap Motor-Actuator and Drive System
Figure 1

3. Flap Controls

A. Removal

(1) To remove link “E”: (See “Figure 2” on page 27505.)

(a) Remove bolt, washer, and nut that secures link “E” to the arm assembly and pushrod assembly.

(b) Remove bolt, nut, washers, and two bushings that secures link “E” to the pushrod and the motor-actuator unit.

(c) Remove safety wire from bolt that secures link “E” to its support bracket. Remove bolt, nut, washers, and cotter pin.

(d) Remove nut, washer and pin securing cable to “E”, and screw, clamp, washer and nut securing cable to cable support or remove nut, four washers and bolt securing potentiometer to “E”.

(e) Remove link “E”.

(2) To remove the 85097-2 pushrod assembly: (See “Figure 2” on page 27505.)

(a) Remove bolt, washer, and nut connecting the pushrod assembly to the arm assembly and link “E.”

(b) Remove bolt, washers, and nut connecting the pushrod assembly to the cam assembly.

(c) Remove the pushrod assembly.

(3) To remove arm assembly: (See “Figure 2” on page 27505.)

(a) Remove bolt, washer, and nut connecting the arm assembly to the pushrod assembly.and link “E”.

(b) Remove the bolt, nut, two bushings, and washers that secure the arm assembly to the tube assembly and aft link assembly.

(c) Remove arm assembly.

(4) To remove the aft link assembly: (See “Figure 2” on page 27505.)

(a) Disconnect flap position indicator cable from arm of the aft link assembly by removing the swing keeper.

(b) Remove bolt, nut, two bushings, and washers that secure the aft link assembly to the tube assembly and arm assembly.

(c) Remove safety wire from bolt that secures the aft link to its support bracket. Remove the bolt, nut, washers, and and cotter pin.

(d) Remove link.

(5) To remove the flap drive pushrod assembly: (See “Figure 2” on page 27505 and “Figure 3” on page 27506.)

(a) Loosen appropriate hose clamp on small end of bellows.

(b) For left flap drive pushrod:

1 Remove bolt, nut, washers, and two bushings that secure the pushrod clevis to link “E” and the actuator-motor assembly.

2 Remove bolt, washer and nut that secures the pushrod clevis to the left inboard bellcrank assembly.

(c) For right flap drive pushrod

1 Remove bolt, two bushings, washers, and nut securing pushrod clevis to the tube assembly, the aft link assembly, and the arm assembly.

2 Remove bolt, washer and nut that secures the pushrod clevis to the right inboard bellcrank assembly.

(d) Slide pushrod from flap drive housing and remove.

(6) To remove the inboard bellcrank: (See “Figure 3” on page 27506.)
   (a) Remove the bolt, washer and nut that secures the bellcrank to the flap drive pushrod.
   (b) Remove nut, bolt, washers and pin that secures the inboard bellcrank to the interconnect pushrod assembly.
   (c) Remove bolt, nut, and washer that secures the bellcrank to the 82913-2 pushrod.
   (d) Remove the bolt, nut, bushing, and pin that secures the bellcrank to its support bracket.
   (e) Remove bellcrank.

(7) To remove the 82913-2 inboard flap actuator pushrod assembly: (See “Figure 3” on page 27506.)
   (a) Remove bolt, nut, and washer that secures the pushrod to the bellcrank.
   (b) Remove bolt, nut, and two washers, that secures the pushrod to the support bracket on the flap.
   (c) Remove the pushrod.

(8) To remove the flap interconnect pushrod assembly: (See “Figure 3” on page 27506.)
   (a) Remove nut, bolt, washers and pin that secures the interconnect pushrod assembly to the inboard bellcrank.
   (b) Remove nut, bolt, two washers, pin and two bushings that secures the pushrod to the 82977-2 pushrod assembly.
   (c) Remove the pushrod.

(9) To remove the idler: (See “Figure 3” on page 27506.)
   (a) Remove nut, bolt, two washers, pin and two bushings that secures idler arm assembly to the interconnect pushrod assembly and the 82977-2 pushrod assembly clevis.
   (b) Remove nut, bolt, two washers, pin and bushing that secures the pushrod to its support bracket.
   (c) Remove the idler arm assembly.

(10) To remove the 82977-2 pushrod assembly: (See “Figure 3” on page 27506)
   (a) Remove, bolt, two washers, pin and two bushings that secures 82977-2 pushrod assembly clevis to the idler arm assembly.
   (b) Remove the bolt, nut, and washer, that secures the pushrod to the outboard bellcrank assembly.
   (c) Remove the 82977-2 pushrod assembly.

(11) To remove outboard bellcrank assembly: (See “Figure 3” on page 27506.)
   (a) Remove bolt, nut, and washer, that secures the 82977-2 pushrod assembly to the outboard bellcrank assembly.
   (b) Remove the bolt, nut, and washer, that secures the 82913-3 pushrod assembly actuator to the outboard bellcrank assembly.
   (c) Remove bolt, nut, two washers, bushing, and cotter pin that secures the outboard bellcrank assembly to the support bracket on the flap.
   (d) Remove the outboard bellcrank assembly.

(12) To remove outboard actuator pushrod assembly: (See “Figure 3” on page 27506.)
   (a) Remove bolt, nut, and washer, that secures the pushrod assembly to the outboard bellcrank arm.
   (b) Remove bolt, nut, two bushings, and two washers, that secures the pushrod to the support bracket on the flap.
   (c) Remove the pushrod.

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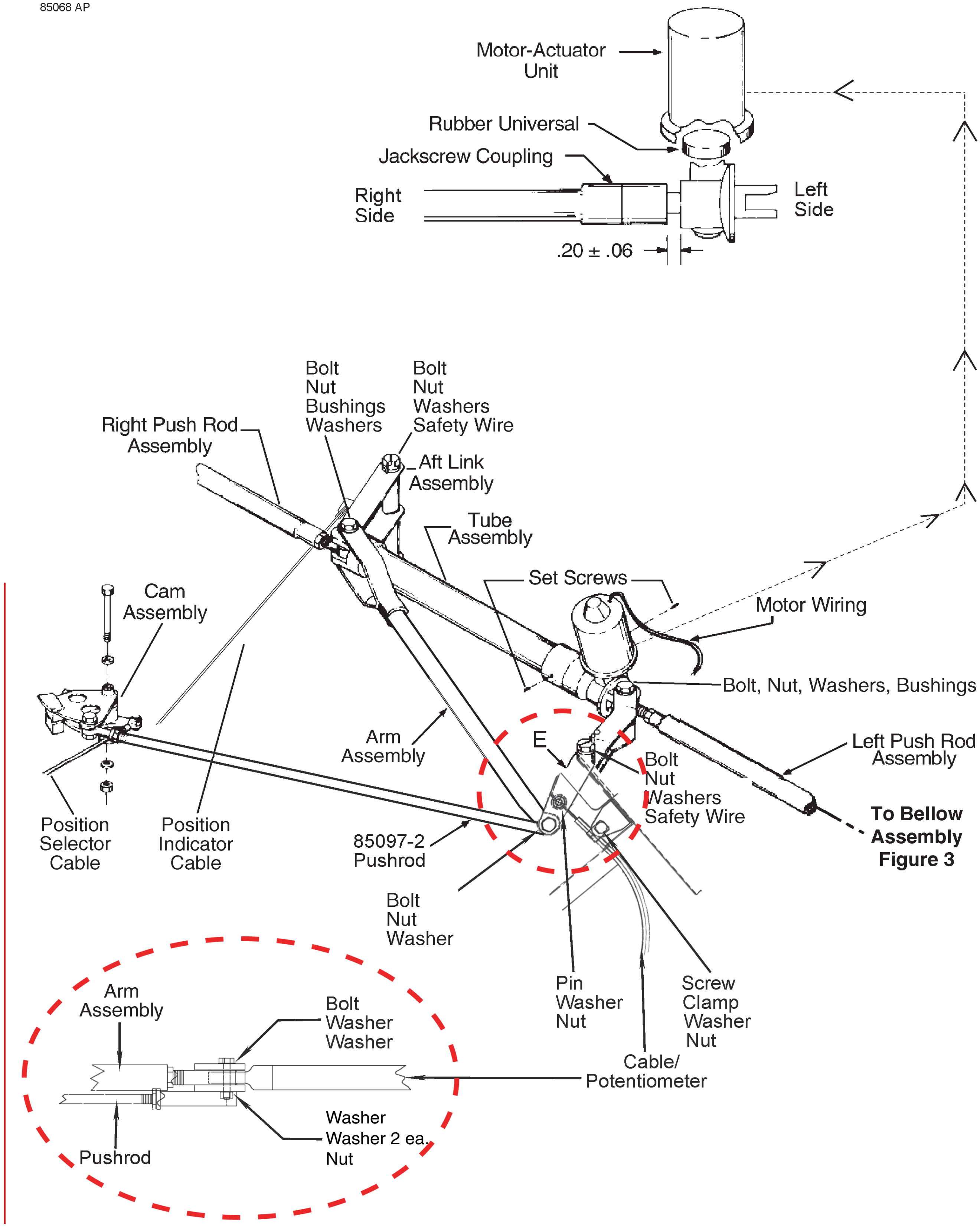
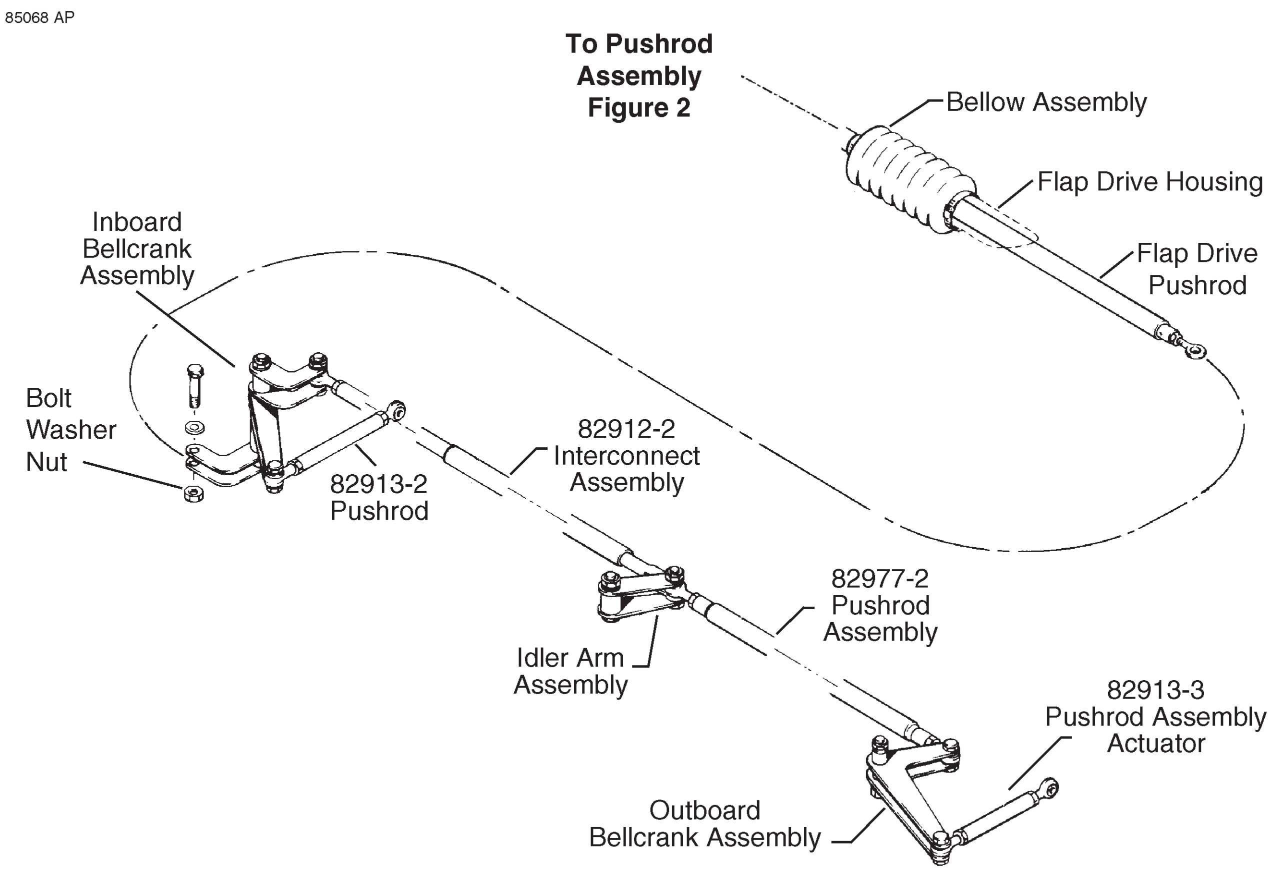


Flap Motor-Actuator and Drive System - Fuselage Components
Figure 2



Flap Motor-Actuator and Drive System - Wing Components
Figure 3

B. Installation

(1) To install outboard actuator pushrod assembly: (See “Figure 3”.)

(a) Ensure that the pushrod assembly measures 4.53 inches between the centers of the rod end holes. (Rod ends should be evenly spaced and locknuts loose)

(b) Attach rod end of pushrod assembly to outboard bellcrank arm with bolt, nut, and washer.

(c) Position pushrod’s aft rod end bearing on flap support. Install bolt (with head pointing outboard), two bushings, two washers (under nut), and nut.

(2) To install the outboard bellcrank: (See “Figure 3”.)

(a) Position bellcrank in support.

(b) Install bolt (head pointing down), bushing, two washers (under nut), and nut. Tighten nut until support arms seat on bushing, then tighten one to two more castellations. Ensure that bellcrank operates freely and install MS24665-134 cotter pin.

(c) Check for a maximum of 0.035 inch vertical end play of bellcrank. Washers may be added as required between the bearing surfaces of the bellcrank and the bellcrank lower supporting bracket, maintaining bellcrank freedom of rotation after pushrod installation. Verify installation of cotter pin.

(d) Attach the 82913-3 pushrod assembly actuator to the outboard bellcrank assembly. Install bolt, nut, and washer.

(e) Attach the 82977-2 pushrod assembly to the outboard bellcrank assembly by installing a bolt, nut, and washer.

(3) To install the 82977-002 pushrod assembly: (See “Figure 3”.)

(a) Ensure that the pushrod assembly measures 24.70 inches between the centers of the rod end holes.

(b) Attach outboard end of the pushrod to the outboard bellcrank assembly by installing NAS bolt, nut, and washer.

(c) Position flap drive interconnect pushrod end into flap 82977-2 pushrod assembly clevis. Install the two bushings. Position this assembly into the idler arms and secure with bolt (head pointing down), two washers (under nut), and nut. Tighten nut until idler arms seat on bushing, then tighten one to two more castellations. Ensure that idler operates freely, then install cotter pin.

(4) To install the idler: (See “Figure 3”.)

(a) Place idler in position.

(b) Install bolt (head pointing down), bushing, two washers (under nut), and nut. Tighten nut until support arms seat on bushing, then tighten one to two more castellations. Ensure that idler operates freely and install cotter pin.

(c) Check for a minimum of 0.035 inch vertical end play of idler. Washers may be added as required between the bearing surfaces of the idler and the idler lower supporting bracket. Verify installation of cotter pin.

(d) Position flap drive interconnect pushrod end into flap 82977-2 pushrod assembly clevis. Install the two bushings. Position this assembly into the idler arms and secure with bolt (head pointing down), two P washers (under nut), and nut. Tighten nut until idler arms seat on bushing, then tighten one to two more castellations. Ensure that idler operates freely, then install cotter pin.

(5) To install flap interconnect pushrod assembly: (See “Figure 3” on page 27506.)

(a) Ensure that the pushrod measures 32.12 inches between the centers of the rod end hole and clevis hole.

(b) Position flap drive interconnect pushrod end into flap 82977-2 pushrod assembly clevis. Install the two bushings. Position this assembly into the idler arms and secure with bolt (head pointing down), two washers (under nut), and nut. Tighten nut until idler arms seat on bushing, then tighten one to two more castellations. Ensure that idler operates freely, then install cotter pin.

(c) Place remaining rod end of pushrod between arms on inboard bellcrank. Use washers as required (Max. 4 top, 4 bottom permitted), maintaining neutral alignment of bearing within the bellcrank. Secure pushrod with bolt (head pointing down), washer (under nut), and nut. Install cotter pin.

(6) To install the inboard flap 82913-2 actuator pushrod assembly: (See “Figure 3” on page 27506.)

(a) Ensure that pushrod measures 4.86 inches between the centers of the rod end bearings. (Rod ends should be evenly spaced and locknuts loose).

(b) Attach rod end of the pushrod assembly to the outboard bellcrank arm with the bolt, washer, and nut.

(c) Position the pushrod aft rod end bearing on the flap supports. Place one bushing on each side of the bearing. Install bolt (head pointing outboard), two washers (under nut), and nut.

(7) To install the inboard bellcrank: (See “Figure 3” on page 27506.)

(a) Position bellcrank on support bracket.

(b) Install bolt (head pointing down), bushing and nut. Tighten the nut until the bracket arms seat on the bushing, then tighten one to two more castellations.

(c) Check for a maximum of 0.035 inch vertical end play of the bellcrank. As required on top of bellcrank, use washers to reduce end play to a minimum and allow free rotation without any binding. As required on bottom of bellcrank, use washers to reduce end play to a minimum and allow free rotation without any binding. Install cotter pin.

(d) Attach flap interconnect pushrod to upper arms on bellcrank. Use washers as required (Max 4 top, 4 bottom permitted), maintaining neutral alignment of bearing within the bellcrank. Secure pushrod with bolt (head pointing down), one washer (under nut), and nut. Install cotter pin.

(e) Attach inboard flap actuator 82913-2 pushrod to long arms on bellcrank with bolt, washer, and nut.

(f) Attach flap drive pushrod to lower arms on bellcrank with the bolt, washer (under bolt head) and nut.

(8) To install the flap drive pushrod assembly: (See “Figure 2” on page 27505 and “Figure 3” on page 27506.)

(a) Adjust the pushrod(s) being installed to measure as follows (rod end holes - center to center):

- right pushrod = 27.34 inches
- left pushrod = 18.58 inches.

(b) Slide the pushrod through appropriate flap drive housing.

(c) For left flap drive pushrod:

1) Position flap drive pushrod clevis in center of U bracket on actuator-motor assembly. Insert the two bushings. Place U shaped clevis of link “E” over actuator-motor bracket. Secure with bolt, washer (under bolt head), washer (under nut), and nut.

2) Secure other end of pushrod to the left inboard bellcrank assembly with a bolt, washer (under bolt head) and nut.

(d) Attach right flap drive pushrod:

1) Position flap drive pushrod in U shaped bracket on end of tube assembly. Insert the two bushings. Position the U shaped bracket on end of arm assembly over bushings with one each washers between bushings and arm assembly bracket. Secure with the bolt, placing washer under the nut. Install nut.

2) Secure other end of pushrod to the right inboard bellcrank assembly with a bolt, washer (under bolt head) and nut.

(e) Tighten the hose clamp on the small end of the bellows.

(9) To install the aft link assembly: (See “Figure 2” on page 27505.)

(a) Position aft link on mounting brackets. Secure with bolt, two washers (under nut), and nut. Tighten nut finger tight plus one or two castellations. Ensure that the aft link operates freely. Install cotter pin and safety with wire.

(b) Install two bushings in hole in tube assembly. Move aft link arms over hole in tube assembly. Move arm assembly yoke over hole, placing one each washers between yoke and aft link arms. Insert bolt through yoke, arms, and actuator. Secure bolt with washer and nut.

(c) Connect the flap position indicator cable to arm of the aft link assembly using the swing keeper.

(10) To install arm assembly: (See “Figure 2” on page 27505.)

(a) Install two bushings in hole in tube assembly. Move aft link arms over hole in tube assembly. Move arm assembly yoke over hole, placing one each and washers between yoke and aft link arms. Insert bolt through yoke, arms, and actuator. Secure bolt with washer and nut.

(b) Secure the arm assembly to the pushrod assembly.and link “E” with an bolt, washer, and nut.

(11) To install the 85097-2 pushrod assembly: (See “Figure 2” on page 27505.)

(a) Secure the pushrod assembly to the cam assembly with bolt, washers, and a nut.

(b) Secure the pushrod assembly to the arm assembly and link “E” with an bolt, washer, and nut.

(12) To install “E” link: (See “Figure 2” on page 27505.)

(a) Position link “E” between support brackets.

(b) Install bolt, washers (under nut), and nut. Tighten nut finger tight plus one to two castellations. Ensure that the link is free to pivot, then install MS24665-132 cotter pin. Safety with wire.

(c) Secure link “E” to the pushrod and the motor-actuator unit with two bushings, washer (under nut), and washer (under bolt head), bolt, and nut.

(d) Secure link “E” to connecting arm assembly and pushrod assembly with an bolt, washer, and nut.

(e) Install nut, washer and pin securing cable to “E”, and screw, clamp, washer and nut securing cable to cable support or install nut, four washers and bolt securing potentiometer to “E”.

C. Flap Motor-Actuator

See "Figure 2" on page 27505.

(1) Removal

(a) Lower flaps to the full down position

WARNING: PULL ELECTRIC FLAP CIRCUIT BREAKER, DISCONNECT BATTERY, AND CHECK THAT NO EXTERNAL POWER IS CONNECTED TO AIRPLANE IN ORDER TO ENSURE THAT NO ELECTRICAL POWER IS BEING APPLIED TO FLAP SYSTEM.

(b) Disconnect flap actuator motor wiring.

(c) Remove bolt, nut, washers, and two bushings connecting left end of actuator left flap drive pushrod and "E" link.

(d) Remove bolt, nut, two bushings, and washers securing right pushrod clevis to the tube assembly, the aft link assembly, and the arm assembly.

(e) Remove motor-actuator and attached tube from aircraft.

(f) Remove the two set screws from actuator jackscrew coupling.

(g) Unscrew tube from actuator.

NOTE: Motor and actuator remain together as one unit.

(2) Installation

WARNING: PULL ELECTRIC FLAP CIRCUIT BREAKER, DISCONNECT BATTERY, AND CHECK THAT NO EXTERNAL POWER IS CONNECTED TO AIRPLANE IN ORDER TO ENSURE THAT NO ELECTRICAL POWER IS BEING APPLIED TO FLAP SYSTEM.

NOTE: It may be necessary to rotate the flap motor-actuator coupling by hand to extend actuator jackscrew to full down position.

(a) Insert actuator into push tube and secure with the two set screws.

(b) Install actuator-push tube assembly to right pushrod assembly, the aft link assembly, and the arm assembly. Secure with bolt, two bushings, one washer one each washer between bushings and arm assembly bracket, and nut.

(c) Connect the left flap drive pushrod clevis in center of U bracket on actuator-motor assembly. Insert two bushings. Place U shaped clevis on link "E" over actuator-motor bracket. Secure with bolt, washer (under bolt head), washer (under nut), and nut.

(d) Connect flap motor-actuator wiring.

(e) Connect battery terminals.

(f) Check flap rigging and operation (refer to Rigging Flaps).

4. Flap System Rigging

(PIR-PPS50056-2. Rev. Y.)

WARNING: PULL ELECTRIC FLAP CIRCUIT BREAKER, BEFORE BEGINNING.

CAUTION: DO NOT ADJUST PRESET LENGTH OF COMPONENTS UNLESS SPECIFIED BELOW.

A. Before rigging flaps

(1) If necessary, disconnect flap pushrods from wing bellcranks.

(2) Rig flaps before lubricating flap tracks.

(3) Check that the length of the arm assembly ("Figure 2" on page 27505) is 14.92 inches. If necessary to reset, adjust the rod end on arm assembly.

B. Rigging flaps

(1) Check the jack screw set dimension is 0.20 ± 0.06 inches as shown in “Figure 2” on page 27505. If adjustment is necessary, hand turn jack screw by grasping rubber universal on motor shaft.

(2) Check that selector lever is in up (0°) position on instrument panel.

(3) Position limit switches at null point by adjusting clevis end of selector cable.

(4) Set circuit breaker. Motor should not run. Move selector lever to 10° position. Motor should run and extend jack screw. Return to up position. Jack screw should return to the 0.20 ± 0.06 inch dimension.

(5) Hold flap trailing edge at approximately W.S. 93.00. Cycle flaps by hand through full travel several times. Flap rollers should operate smoothly in the track without a tendency to grab or bind.

(6) Keep flap rollers firmly bottomed out in forward end of center and inboard flap track slots. Attach flap rods to wing bellcranks. Outboard flap rod is to be snug, allowing no flap end play. Inboard rod is to be snugged and then tightened one half turn.

(7) Move selector lever to 10° position. Verify flap position is as specified in “Chart 2” on page 27004. Right and left flap shall be within one (1) degree of each other. Check travel using suitable tool.

(8) Move selector lever to 20° position. Verify flap position per “Chart 2” on page 27004. Right and left flap shall be within one (1) degree of each other. Check travel using suitable tool.

(9) Move selector lever to Full flaps down position. Verify flap position is as specified in “Chart 2” on page 27004. Right and left flap shall be within one (1) degree of each other. Check travel using suitable tool.

NOTE: Maintain a slight up pressure on the underside of flap trailing edge when checking the 10°, 20° and Full flaps down position.

(10) Move selector lever to the Full up position. Verify flap position per “Chart 2” on page 27004.

NOTE: Maintain a slight up pressure on the underside of flap trailing edge when checking this position.

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF FLAPS. UPON COMPLETION OF FLAP RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE FLAP TRAILING EDGES MOVE UP WHEN THE SELECTOR LEVER IS UP; AND, THAT THE FLAP TRAILING EDGES MOVE DOWN WHEN THE SELECTOR LEVER IS DOWN.

(11) Proceed to “Flap System Adjustment”, below.

5. Flap System Adjustment

Use the following procedure to adjust flap system:

A. To align flaps within one (1) degree of each other, adjust the inboard rod end on either the left or right pushrod assembly (i.e., 82977-2, “Figure 3” on page 27506).

NOTE: Maintain a slight up pressure on the underside of the flaps when checking flap travel.

B. The outboard rod end of the pushrod 85097-2 (see “Figure 2” on page 27505) bolts to a slot in the cam assembly. To increase flap travel, move that rod end forward in the slot. To decrease flap travel, move that rod end aft in the slot. Adjust rod end as required to allow the cam assembly to activate the warning horn switch.

NOTE: It is acceptable to have up to 3/16th inch vertical end play at flap trailing edge in the 0° flap position, measured at center of flap track.

C. Proceed to “Post-Rigging Inspection” on page 275012.

6. Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any flap actuator, motor, screw jack, drive cable, pushrod, or torque tube replacement, disconnect, or rigging perform the following post-rigging inspection:

A. Place the flap control lever in the DOWN position. Visually confirm both flaps move fully DOWN when electrical power is applied and remain ALIGNED. That is the trailing edge of each flap is significantly LOWER than when streamlined with the wing.

B. Place the flap control lever in the UP position. Visually confirm both flaps move fully UP when electrical power is applied and remain ALIGNED. That is the trailing edge of each flap is streamlined with the wing.

C. Make a logbook entry documenting completion of this inspection.

7. Flap Position Indicator and Flap Warning System Functional Test

S/N's 4636001–4636459, 4636461–4636462, 4636481; and 4692001–4692133, 4692141, 4692149, 4692153 only.

NOTE: To verify flap position indication on G1000/G1000 NXi equipped aircraft, see Flaps Calibration under LRU Test Procedures in 34-25-01 (G1000) or 34-25-02 (G1000 NXi).

A. Rig the flaps as described above.

B. Place the airplane on jacks and fully retract the landing gear.

C. Place flap handle in 10° position. Flaps shall extend to first mid position and landing gear warning horn does not sound.

D. Place flap handle in 20° position. Flaps shall extend to second mid position and warning horn sounds.

E. Place flap handle in full down position. Flaps shall extend fully, and warning horn sounds.

F. Place flap handle in full up position. Flaps shall travel to up position and warning horn should stop sounding.

G. Extend landing gear until fully down and locked. Remove airplane from jacks.

H. Move flap handle thru 10°, 20° and full down position, the warning horn should not sound. Check the flap indicator (not applicable to S/N's 4692001–4692133, 4692141, 4692149, 4692153) to confirm the indicator correctly follows the flap handle and correctly displays the appropriate flap position.

8. Flap Warning Light/Message Functional Test

A. In non-G1000 aircraft only:

Verify FLAPS annunciator lamp is good by activating "Press to Test" button on annunciator panel. Lamp should illuminate.

B. Pull FLAP MOTOR circuit breaker.

C. Check FLAP WARN breaker is set.

D. With power on, momentarily short terminals on current sensor (see "Figure 4") until FLAPS light illuminates (in non-G1000 aircraft), or FLAP FAIL crew alert system message annunciates (in G1000/G1000 NXi equipped aircraft). Light/message should illuminate within 5 seconds.

E. Remove jumper.

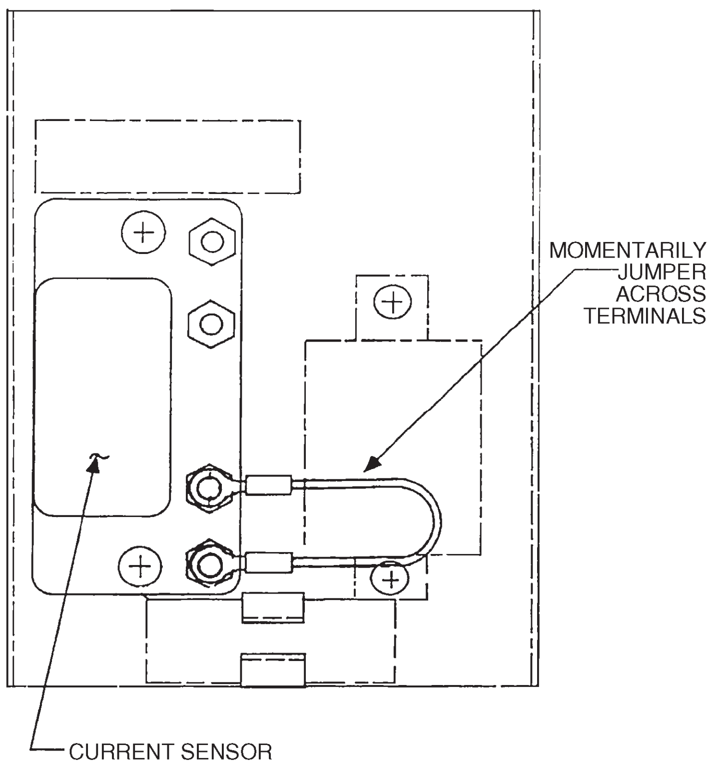
F. Set FLAP MOTOR circuit breaker.

G. Set flap lever to any down position - flaps should not operate, light/message remains on.

H. Return flap lever to full up position.

I. Pull and reset FLAP WARN circuit breaker, light/message should go out.

J. Operate flap lever - operation should be normal.



Flap Control Assembly Current Sensor
Figure 4

9. Flap Bellcrank Distortion Inspection

If flaps have been extended at or above V_{FE}, inspect the flap bellcranks for evidence of distortion.

A. Access the flap bellcrank through the main landing gear wheel well. The bellcrank is outboard the wing root rib at W.S. 29.00 and adjacent to the aft spar.

B. If the paint is cracked or peeling anywhere along the bellcrank (see “Figure 5”), torsional movement may have occurred.

C. Remove the paint with a commercially available paint stripper, such as Cee-Bee E2012 or E2060.

D. Using a dye penetrant method of inspection per latest revision of AC 43.13-1.

   (1) Inspect for cracks at either arm on the bellcrank shaft near the welds.

   (2) Inspect for cracks on the shaft itself as shown in “Figure 5”.

E. If no cracks are found, repaint the bellcrank(s) and reinstall.

F. If cracks are found:

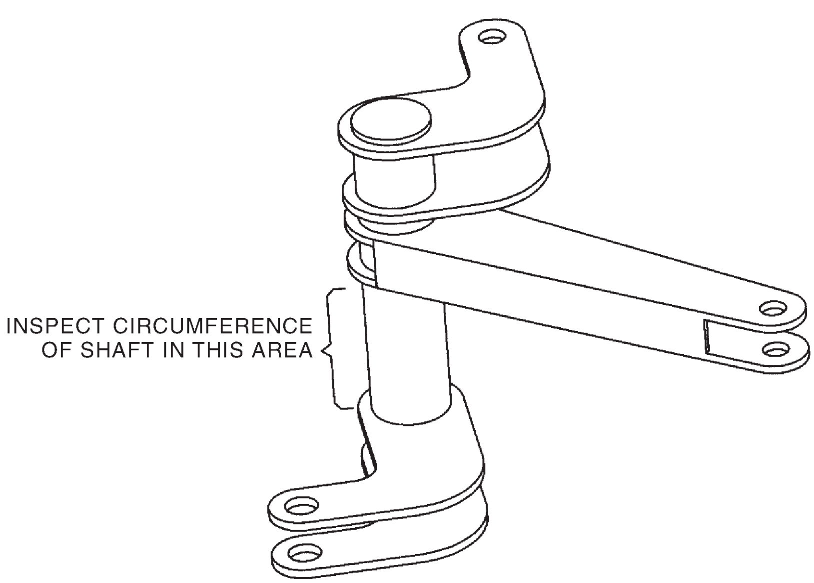
   (1) Replace the cracked bellcrank with a new part as follows:

      (a) Lefthand side = 82905-010 bellcrank assembly.

      (b) Righthand side = 82905-011 bellcrank assembly.

   (2) Inspect the rest of the flap system for distortion or damage to the extension rods and rod ends.

G. Make a logbook entry.



Flap Bellcrank Distortion Inspection
Figure 5

SPEEDBRAKES

WARNING: DO NOT PUT FINGERS IN HOLES OF EXTENDED SPEEDBRAKE BLADES. IF POWER TO THE SYSTEM IS INTERRUPTED, THE SPEEDBRAKE BLADES WILL CLOSE WITH SUFFICIENT FORCE TO AMPUTATE FINGERS.

1. Description and Operation

A Precise Flight SpeedBrake 2000 System is available as an option on S/N's 4636187 and up, and S/N's 4692001 and up. When installed, the system provides expedited descents at low cruise power, glide path control on final approach, and an aid to the prevention of engine shock cooling.

The system consists of wing-mounted electric SpeedBrake cartridges, an asymmetric logic switching unit, an annunciator light, a 3-amp circuit breaker, and a yoke-mounted actuator switch. Operation is simple ON or OFF. When power is supplied to the cartridges, the blades extend; when power is interrupted, powerful springs retract the blades.

CHART 1
TROUBLESHOOTING SPEEDBRAKE SYSTEM

Trouble	Cause	Remedy
SpeedBrake fails to retract, and pulling circuit breaker returns brakes to stowed.	Retract Microswitch fails to operate.	Replace or repair Microswitch - if Microswitch is OK, use schematic to find break in wiring.
	Switch inoperative.	Replace or repair switch - if switch OK use schematic to find break in circuit.
	Motor and Asymmetric Logic Control Unit failed.	Replace Asymmetric Logic Control Unit.
CAUTION: DO NOT PUSH SPEEDBRAKE BLADES DOWN, IF THEY FAIL TO RETRACT. FORCING SPEEDBRAKE BLADES DOWN WILL DAMAGE THE UNIT.		
SpeedBrake fails to retract, and pulling circuit breaker fails to return the brakes to stowed.	Clutch failure.	Pull brakes apart slightly, if this does not provide remedy replace clutch assy.
	Spring failure.	Replace spring.
SpeedBrake fails to extend NOTE: If one SpeedBrake fails to extend, the Logic Interconnect Box will cut clutch power to the other SpeedBrake Cartridge after 4 seconds - the cartridge motor, however will continue to run.	Extend Microswitch fails to operate.	Replace or repair Microswitch - if Microswitch is OK, use schematic to find break in wiring.
	Clutch inoperative.	Replace Clutch Assembly.
	Switch inoperative.	Replace or repair switch - if switch OK, use schematic to find break in circuit.
	Motor and Asymmetric Logic Control failed.	Replace Asymmetric Logic Control Unit.

2. Troubleshooting

See "Chart 1" on page 27601.

3. Inspections

A. 50 Hour Inspection

(See "Figure 1".)

(1) Inspect Cap Strip Cover Screws and Top Attachment Screws for security. If loose, remove screws and reinstall using Locktite 242. Torque to eight (8) in. lbs.

(2) Inspect drain tube for obstructions or debris.

B. Annual Inspection

(1) Remove SpeedBrake cartridge as described under SpeedBrake Cartridges, Removal, below.

(2) Clean and inspect unit for damage, corrosion, looseness, and proper operation.

(3) Lubricate worm and worm gear with Aeroshell 22.

(4) Reinstall SpeedBrake cartridge as described under SpeedBrake Cartridges, Installation, below.

4. SpeedBrake Cartridges

WARNING: DO NOT PUT FINGERS IN HOLES OF EXTENDED SPEEDBRAKE BLADES. IF POWER TO THE SYSTEM IS INTERRUPTED, THE SPEEDBRAKE BLADES WILL CLOSE WITH SUFFICIENT FORCE TO AMPUTATE FINGERS.

CAUTION: DO NOT PUSH SPEEDBRAKE BLADES DOWN, IF THEY FAIL TO RETRACT. FORCING SPEEDBRAKE BLADES DOWN WILL DAMAGE THE UNIT.

The SpeedBrake cartridges (which are in each wing between W.S. 93.00 and 107.50) consist of a gearmotor and clutch assembly driving the interconnected blades through a worm gear set. The blades are loaded to the retracted position by a constant force spring.

The SpeedBrake cartridge also contains two microswitches; one for the retracted position and one for the extended position. The microswitches control gearmotor power, gearmotor direction, signals the cockpit annunciator and signals the logic-switching unit.

A. Removal

(1) Remove access plates and covers.

NOTE: If installed, the radome on the right wing will need to be removed first.

(2) Disconnect wiring harness.

(3) Pull the drain tube up through the wing skin.

(4) Remove the spring clip on each side of the SpeedBrake cartridge and remove the cartridge.

B. Installation

NOTE: In S/N's 4636001–4636715 and 4636717–4636719, SpeedBrake cartridge units must be installed as a sequentially serialized pair as received from the factory.

(1) Insert the SpeedBrake cartridge through the access hole ensuring that the spring clip hinge halves are on the aft side of the cartridge.

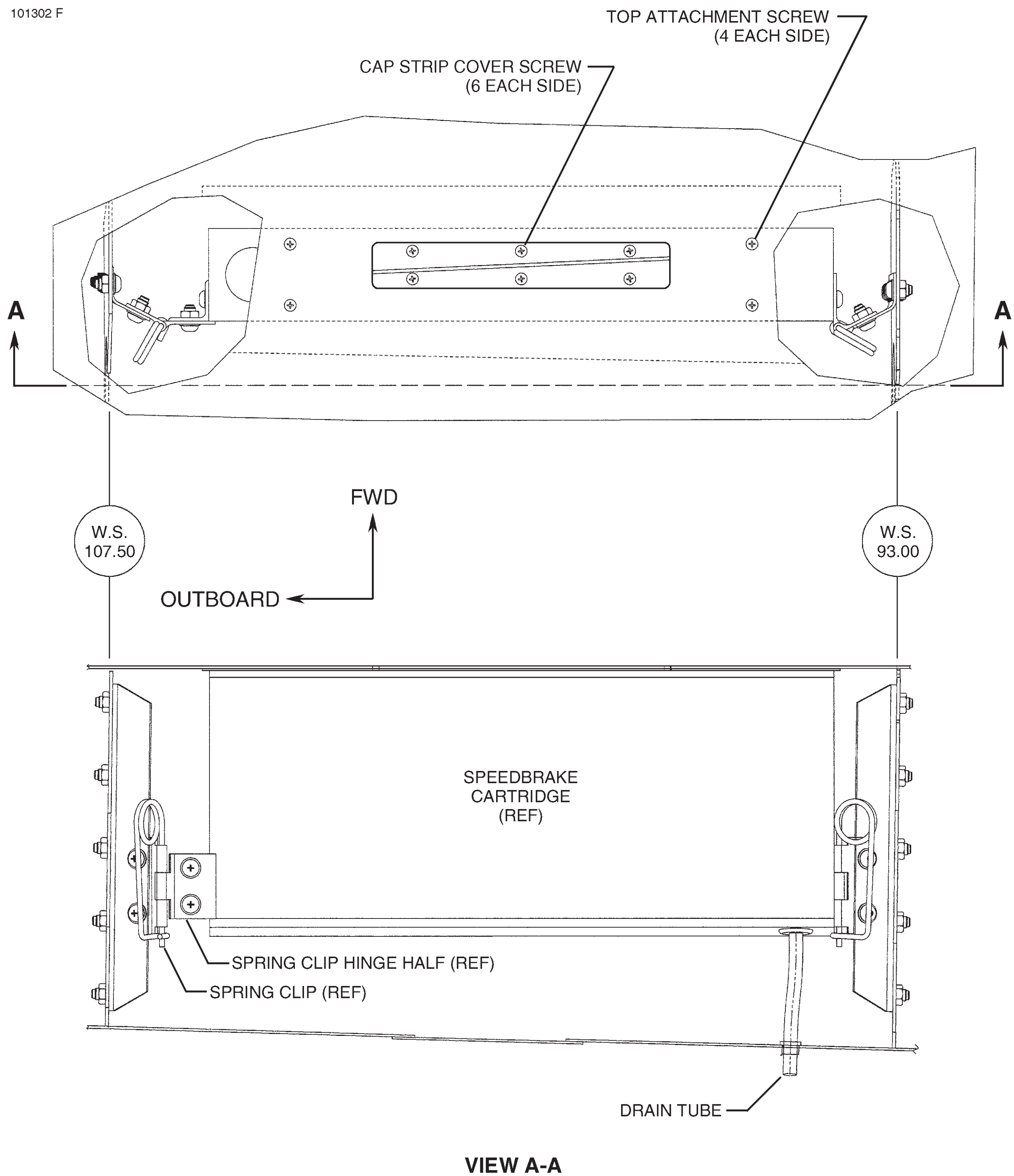
(2) Position the cartridge aligned with and between the spring clip hinge halves on the mounting brackets. Secure with spring clips.

(3) Feed the drain tube through the wing skin.

(4) Connect the wiring harness.

(5) Reinstall access plates and covers.

NOTE: If required, reinstall the radome on the right wing.



SpeedBrake 50 Hour Inspection
Figure 1

Effectivity
4636187 and up,
and PA-46R-350T, if Installed

5. Asymmetric Logic Switching

The Asymmetric Logic Switching Unit (which is located under the rear co-pilot floor closeout) monitors SpeedBrake cartridge position and will cut power to both SpeedBrake cartridge clutches, causing instant blade retraction. This happens for two conditions:

A. Both SpeedBrake cartridges fail to extend completely within four (4) seconds, or

B. One SpeedBrake cartridge is contacted after extension (birdstrike, large hail etc).

NOTE: The gearmotors will continue to run after clutch power has been disconnected, with the SpeedBrake switch in the ON position.

6. SpeedBrake Switch

The control yoke-mounted SpeedBrake switch controls SpeedBrake extension and retraction.

7. SpeedBrake Annunciator

The SpeedBrake annunciator (if installed) is mounted (separately) on the instrument panel above the pilot's airspeed indicator. It illuminates after both SpeedBrakes are extended and extinguishes after both SpeedBrakes are stowed.

NOTE: On G1000/G1000 NXi equipped aircraft, the annunciation is incorporated into the PFD/MFD. See Crew Alerting System (CAS) information in 31-50-00.

8. Testing

A. Basic Functional Test

A properly operating SpeedBrake system should function as specified below:

(1) Turn on aircraft power.

(2) Switch SpeedBrakes ON. Full deployment should occur in no more than 2.5 seconds.

WARNING: DO NOT PUT FINGERS IN HOLES OF EXTENDED SPEEDBRAKE BLADES. IF POWER TO THE SYSTEM IS INTERRUPTED, THE SPEEDBRAKE BLADES WILL CLOSE WITH SUFFICIENT FORCE TO AMPUTATE FINGERS.

(3) Verify motor stop. Verify full blade extension. Proper blade position is 87° ± 2°.

CAUTION: DO NOT PUSH SPEEDBRAKE BLADES DOWN, IF THEY FAIL TO RETRACT. FORCING SPEEDBRAKE BLADES DOWN WILL DAMAGE THE UNIT.

(4) Switch SpeedBrakes OFF. Full retraction should occur in no more than 2.0 seconds.

(5) Verify motor stop. Verify full blade retraction. Proper blade position is 0° ± 2°.

(6) Repeat steps (2) thru (5) several times to verify proper operation.

(7) Turn off aircraft power.

B. Speedbrake Cartridge Tester

Precise Flight offers a SpeedBrake Cartridge Test Connector which is used to conduct a more precise functional test. The cartridge test connector is available from your Piper Distributor as P/N 762-138 and is used as follows:

(1) Turn aircraft power OFF.

(2) Remove left SpeedBrake access panel.

(3) Disconnect left SpeedBrake cartridge from wiring harness.

(4) Connect the cartridge test connector to the left speedbrake wiring harness.

(5) Turn aircraft power ON and operate SpeedBrake system.

WARNING: DO NOT PUT FINGERS IN HOLES OF EXTENDED SPEEDBRAKE BLADES. IF POWER TO THE SYSTEM IS INTERRUPTED, THE SPEEDBRAKE BLADES WILL CLOSE WITH SUFFICIENT FORCE TO AMPUTATE FINGERS.

CAUTION: DO NOT PUSH SPEEDBRAKE BLADES DOWN, IF THEY FAIL TO RETRACT. FORCING SPEEDBRAKE BLADES DOWN WILL DAMAGE THE UNIT.

(6) With the right SpeedBrake extended, press on the extended blades to simulate air loading.

(a) If the blades drop, turn aircraft power OFF, replace the right SpeedBrake, and start with step (5), above.

(b) If the blades remain extended, turn aircraft power OFF, remove the cartridge test connector from the left SpeedBrake wiring harness, and reconnect the wiring harness to the left SpeedBrake cartridge.

(7) Turn aircraft power ON and operate SpeedBrake system.

WARNING: DO NOT PUT FINGERS IN HOLES OF EXTENDED SPEEDBRAKE BLADES. IF POWER TO THE SYSTEM IS INTERRUPTED, THE SPEEDBRAKE BLADES WILL CLOSE WITH SUFFICIENT FORCE TO AMPUTATE FINGERS.

CAUTION: DO NOT PUSH SPEEDBRAKE BLADES DOWN, IF THEY FAIL TO RETRACT. FORCING SPEEDBRAKE BLADES DOWN WILL DAMAGE THE UNIT.

(8) Press on the extended blades of the left SpeedBrake to simulate air loading.

(9) If the blades on both the left and right SpeedBrakes drop, turn aircraft power OFF, replace the left SpeedBrake, and start with step (7), above.

(10) When the blades on both SpeedBrakes remain extended, the test has been completed successfully.

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CHAPTER

28

FUEL

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CHAPTER 28

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
28-List of Effective Pages	1	Mar 25/21
	2	Sep 15/09
28-Table of Contents	1	Nov 30/17
	2	Mar 25/21
28-00-00	1	Nov 30/17
	2	Sep 15/09
	3	Nov 30/17
	4	Sep 15/09
	5	Sep 15/09
	6	Sep 15/09
28-10-00	1	Dec 15/14
	2	Sep 15/09
	3	Sep 15/09
	4	Sep 15/09
	5	Sep 15/09
	6	Nov 30/17
	7	May 31/11
	8	Sep 15/09
28-20-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Sep 15/09
	5	Sep 15/09
	6	Nov 30/17
28-40-00	1	Mar 25/21
	2	Mar 25/21
	3	Mar 25/21
	4	Mar 25/21
	5	Mar 25/21
	6	Mar 25/21
	7	Mar 25/21
	8	Mar 25/21
	9	Mar 25/21
	10	Mar 25/21
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	16	Mar 25/21
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	19	Mar 25/21
	20	Mar 25/21
	21	Mar 25/21
	22	Mar 25/21
	23	Mar 25/21
	24	Mar 25/21

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CHAPTER 28 - FUEL

TABLE OF CONTENTS

CHAPTER 28 - FUEL

TABLE OF CONTENTS (continued)

GENERAL

WARNING: DURING ANY FUEL SYSTEM MAINTENANCE, USE A GROUND WIRE ATTACHED FROM EACH MAIN GEAR GROUNDING PIN TO SEPARATE APPROVED GROUNDING STAKES TO PREVENT THE ACCIDENTAL DISCONNECTION OF AIRPLANE GROUNDING.

1. Fuel System

A. Description

The fuel system consists of an integral fuel tank in each outboard wing section, having a usable capacity of 60 U.S. gallons for a total usable capacity of 120 U.S. gallons. The fuel tanks are part of the wing structure - i. e. - a wet-wing configuration. Each tank has an anti-icing fuel vent and forward and aft fuel supply lines with finger screens. Fuel flows from these finger screens through lines, to a one (1) U.S. gallon collector tank located in each wing wheel well area. Installed in each collector tank is an electrically-driven centrifugal fuel boost pump, which is activated in conjunction with the fuel selector control in the cockpit. A sump drain is also located in each collector tank.

Fuel leaving the left or right collector/sump tank flows to a selector valve which is located on the right fuselage side behind the copilot's seat in a non-pressurized compartment. All fuel lines passing through the pressurized cabin are metal tubes surrounded by plastic cushion and encased by a second metal tube. This second tube is sealed from the cabin environment to preclude fuel from entering the cabin area or pressurized cabin air from entering fuel lines in the event of a leak.

The selector valve is cable-controlled, by a thumb-sized handle (just below the fuel quantity gauges), with detents at the OFF, LEFT, and RIGHT positions. Selecting LEFT or RIGHT directs fuel flow to the engine from the tank selected. To select OFF, the fuel selector must be moved to the left tank position, moved down against spring pressure, and then moved to the far left, or OFF position.

Fuel flows from the fuel selector valve forward to the fuel filter located below the baggage floor on the right side. The filter drain is a nylon tube located on the right side of the aircraft, forward of the wing. Fuel flows from the filter, forward through the emergency fuel pump and firewall, into the engine compartment, and then to the engine-driven fuel pump.

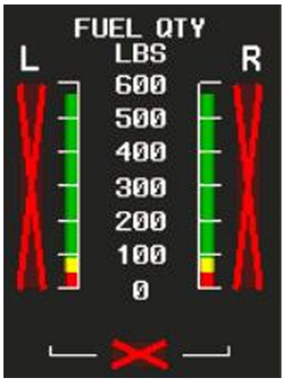
B. Troubleshooting

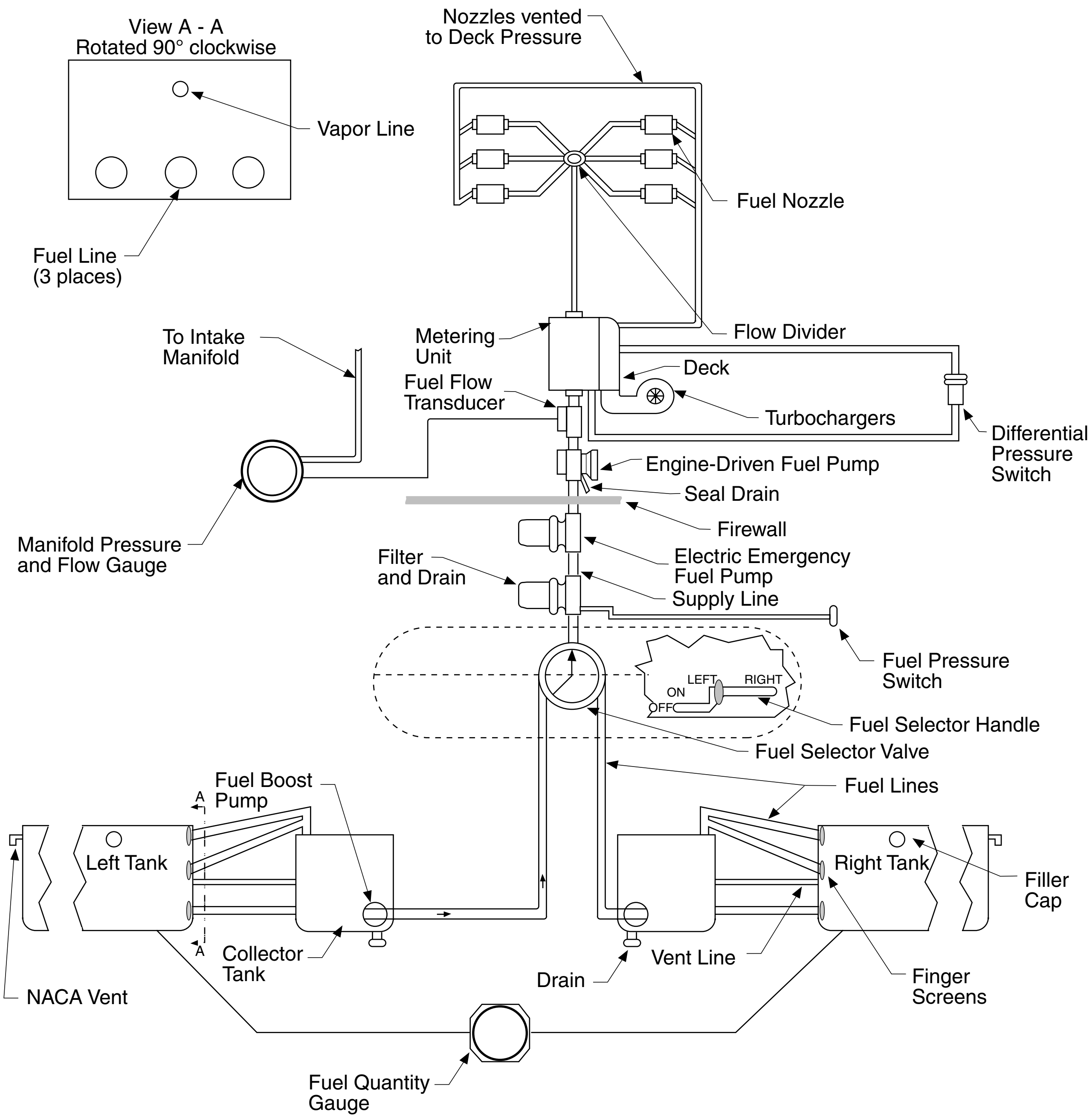
"Chart 1" on page 28002 lists troubles which may occur in the mechanical or electrical portions of the fuel system, the probable cause, and suggested remedy. "Chart 2" on page 28003 lists troubles unique to G1000/G1000 NXi equipped aircraft. When troubleshooting, first check from the fuel supply or power source to the item affected.

CHART 1
TROUBLESHOOTING FUEL SYSTEM

Trouble	Cause	Remedy
Failure of fuel to flow.	Fuel line blocked.	Flush fuel system.
	Fuel vent cap blocked.	Check and clean vent hole in cap.
	Mechanical or electrical fuel pump failure.	Check and replace if necessary.
	Fuel selector valve in improper position.	Reposition as required.
		Check for obstructions in the fuel selector leverage mechanism.
	Damaged fuel selector valve.	Replace fuel selector valve.
Fuel quantity gauge fails to operate.	Broken wire.	Check and repair.
	Gauge inoperative.	Replace gauge.
	Fuel sender float partially or completely filled with fuel.	Replace sender.
	Circuit breaker open.	Check and reset.
	Float and arm assembly of fuel sender sticking.	Check.
	Bad ground.	Check for good contact at ground lip or rear of gauge.
No fuel pressure indication.	Fuel selector valve stuck.	Check fuel selector valve.
	Fuel tanks empty.	Check fuel tanks and fill.
	Defective gauge.	Replace gauge.
	Fuel selector valve in improper position.	Reposition fuel selector valve lever.
Lower pressure or pressure surges.	Obstruction in inlet side of pump.	Trace lines and locate obstruction.
	Air in line to pressure gauge.	Bleed line.

CHART 2
TROUBLESHOOTING FUEL SYSTEM - G1000/G1000 NXi

Trouble	Cause	Remedy
Fuel quantity display has red X's L & R	Circuit breaker open.	Check and reset.
 G1000 Shown	Faulty Fuel Quantity Probes or GEA harness.	Check and repair.
	Faulty Fuel Quantity Probes, GEA, or #1 PFD.	Check, Test, Replace as required.
	Faulty fuel sender(s).	Replace sender(s).
	Faulty fuel sender harness.	Check and repair.



Fuel System
Figure 1

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STORAGE

The wing tanks are an integral part of the outboard wing sections. The liquid tight wing tanks located between W.S. 116.00 and W.S. 240.00 are formed by bonding the wing skins with the ribs and stringers (i.e. - wet-wings). Holes in the ribs and spar permit movement of the fuel within the wings. There are ten access covers on the underside of each wing for access to the sealed portion of the tank.

1. Access Covers

NOTE: In S/N's 4636001 thru 4636379, if any press fit / clinch nut in an access cover retainer becomes loose and spins, it may be replaced with a riveted nutplate as follows:

Press Fit / Clinch Nut	replaced with	Nutplate	and	Rivets
PS10062-10-2 (467-511)		MS21059L08 (477-809)		MS20426AD3
PS10062-10-3 (603-312)		NAS1474A08 (477-899)		MS20426AD3

In S/N's 4636001 thru 4636379, wet-wing access covers are sealed with sealant and secured with screws.

In S/N's 4636380 and up and S/N's 4692001 and up, the wet-wing access covers use Mold-in-Place seals and are secured with screws.

NOTE: In S/N's 4636380 and up, the early plain access covers can be used if the newer Mold-in-Place seal access covers are not available, but they then must be installed using sealant as described for S/N's 4636001 thru 4636379 under Installation, below.

A. Removal

(1) Drain all fuel from the wing tank being serviced.

(2) Ensure electrical power is "OFF".

(3) Remove the access cover as follows:

(a) In S/N's 4636001 thru 4636379, remove the screws securing the access cover. With a suitable tool, pry the access cover away from the wing, taking care not to damage the mating surfaces - especially on the wing. When the access cover becomes free, carefully lower the cover and sender out of the wing until the electrical connection can be reached and disconnected.

(b) In S/N's 4636380 and up and S/N's 4692001 and up, remove the screws securing the access cover and carefully lower the cover and sender out of the wing until the electrical connection can be reached and disconnected.

B. Installation

(PIR-PPS50076, Rev. New.)

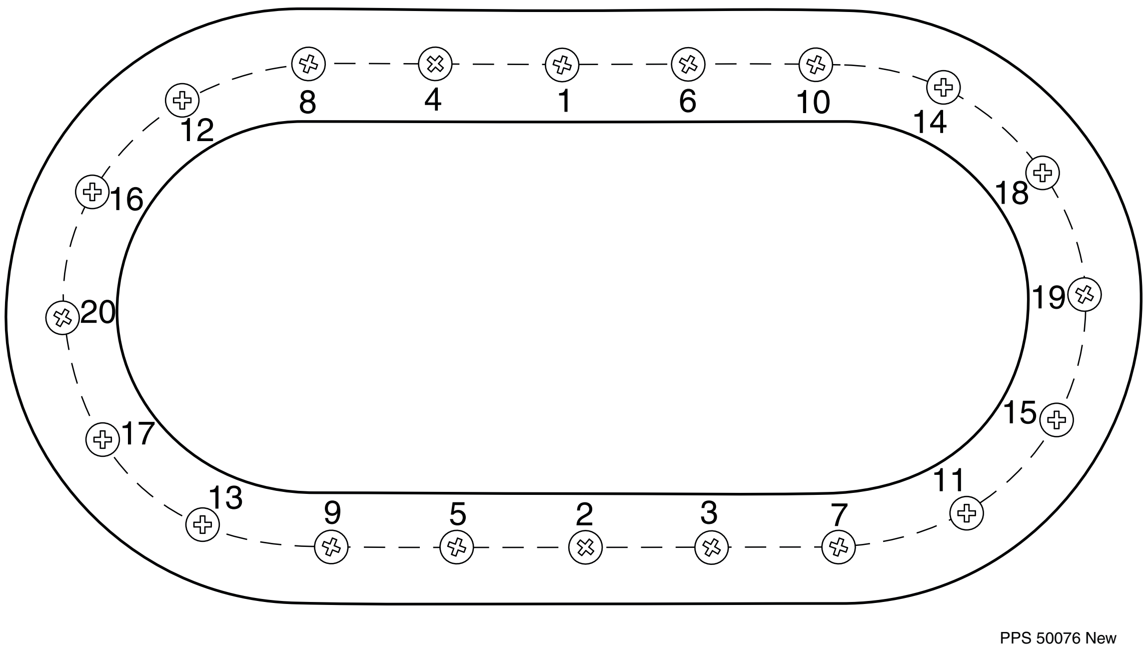
(1) Ensure electrical power is "OFF".

(2) In S/N's 4636001 thru 4636379, apply Products Research Co. (PRC) PR1428-B-2 (or PR1403G-B-2) sealant and install the access cover on the wing with screws. Allow sealant to cure 72 hours before refilling tanks.

CAUTION: DO NOT USE ACETONE TO CLEAN COVER AND MATING SURFACE, AS IT IS NOT COMPATIBLE WITH MOLD-IN-PLACE SEAL MATERIAL.

(3) In S/N's 4636380 and up and S/N's 4692001 and up, clean access cover and mating surface using isopropyl alcohol. Allow to air dry for 15 minutes minimum. Loosely install all screws to hold the panel in place. Using the sequence in "Figure 1", lightly snug all screws. Following the same pattern torque all screws to 25 inch.-lbs.

Final torque all screws to 25 to 30 inch pounds



Effectivity
4636380 and up

Fuel Access Cover Torque Pattern
Figure 1

2. Leak Evaluation

Routine monitoring of the fuel storage system and prompt attention to any anomalies discovered will ensure trouble-free operation. Most fuel leaks will be identified by the pilot during pre-flight inspection and will generally consist of a stain or leak around the vent scoop, filler, drain valves, and wing surface in the wet-wing or along the plumbing lines in the wing or engine compartment. The following provides criteria for evaluating various categories of leaks and the maintenance action required.

A. Wing Tanks.

WARNING: CAREFULLY EXAMINE ANY LEAKAGE TO ENSURE THAT IT IS NOT PROGRESSING TO A CRITICAL AREA OF THE AIRPLANE AND THAT THERE IS NO POSSIBILITY OF FUEL BEING BLOWN INTO THE FUSELAGE AREAS.

(1) Upper wing surface.

The airplane may be released for flight with a fuel leak in the upper wing surface provided the required fuel load will allow the fuel level to remain below the leak area.

CHART 1
WET-WING FUEL LEAK EVALUATION

FUEL TANK LEAKAGE LOCATION	HEAVY STAIN	RUNNING AND SEEP	SEEP	RUN	DRIPPING
Upper Wing Surface	1	1	2	3	4
Rear of Fwd. Spar	1	2	3	3	4
Behind Leading Edge	2	2	3	4	4
Lower Wing Surface	1	2	3	4	4

REQUIRED ACTION

1. No repair action required, frequent inspections must be made to assure that leak does not progress.
2. No immediate repair required. Frequent inspection must be made to assure that leak does not progress. Repair must be made within the next 100 hours.
3. Flight may be continued to base or to repair facility for repair at pilot's discretion.
4. Immediate repair required.

(2) Lower wing surface.

Aircraft may be released for flight with fuel leaks in the lower wing skin as follows: (See "Figure 2") Stain (Detail A), Seep (Detail B), and Heavy Seep (Detail C).

(3) Fuel leakage from the integral fuel tanks (i.e. - wet-wing) can be divided into the five categories shown in "Figure 2". Each category is identified by a visual examination of the wetted area around the source of leakage over time.

To evaluate a leak, clean all wetted area in the vicinity of a leak. Wait two (2) hours and reexamine the leak area and apply the following criteria to identify the leak category and required maintenance action. (See "Figure 2" and "Chart 1" .)

(a) A stain (Detail A) is evaluated as a leak where fuel wets an area around the apparent leak source such that the wetted area width is not over two (2) inches.

(b) A seep (Detail B) is defined as a leak where the fuel wets an area around the apparent leak source such that the wetted area width is not over six (6) inches. The fuel must not run, flow, drip, or have any resemblance to any one of these conditions.

(c) A heavy seep (Detail C) is a leak where fuel appears to spread very slowly and covers an area larger than the six (6) inches specified under seep, but does not flow, run or drip.

(d) A run (Detail D) is defined as a leak where the fuel is dripping at a rate of less than ten (10) drops per minute or flowing slowly. A leak of this type, readily identifiable as fuel, will appear immediately after wiping the area clean.

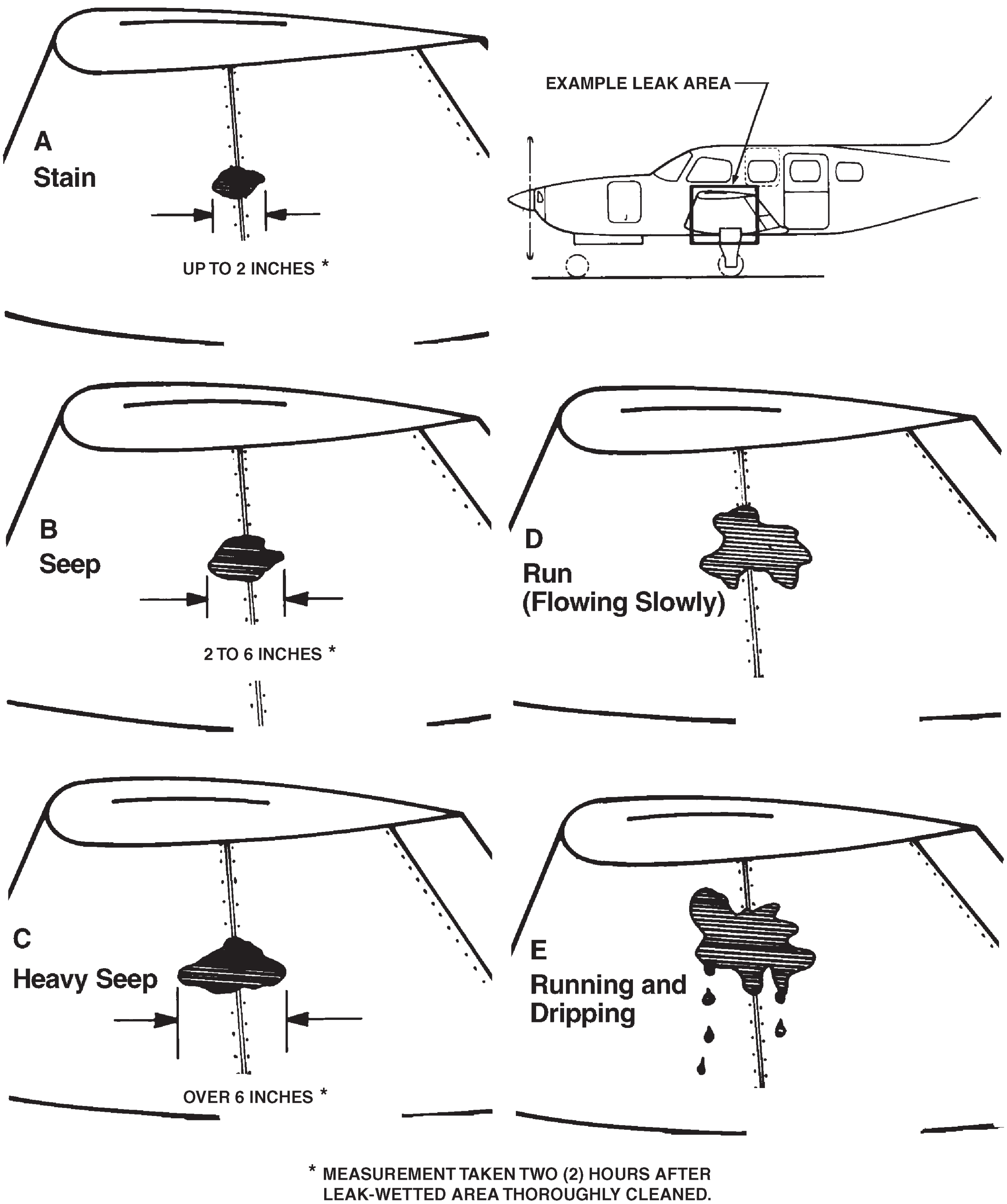
(e) A running or dripping leak (Detail E) is any leak in excess of the run leak (Detail D). This type requires immediate repair of the leak.

NOTE: These five (5) categories are based on re-examination of the leak two (2) hours after the wetted area has been thoroughly cleaned.

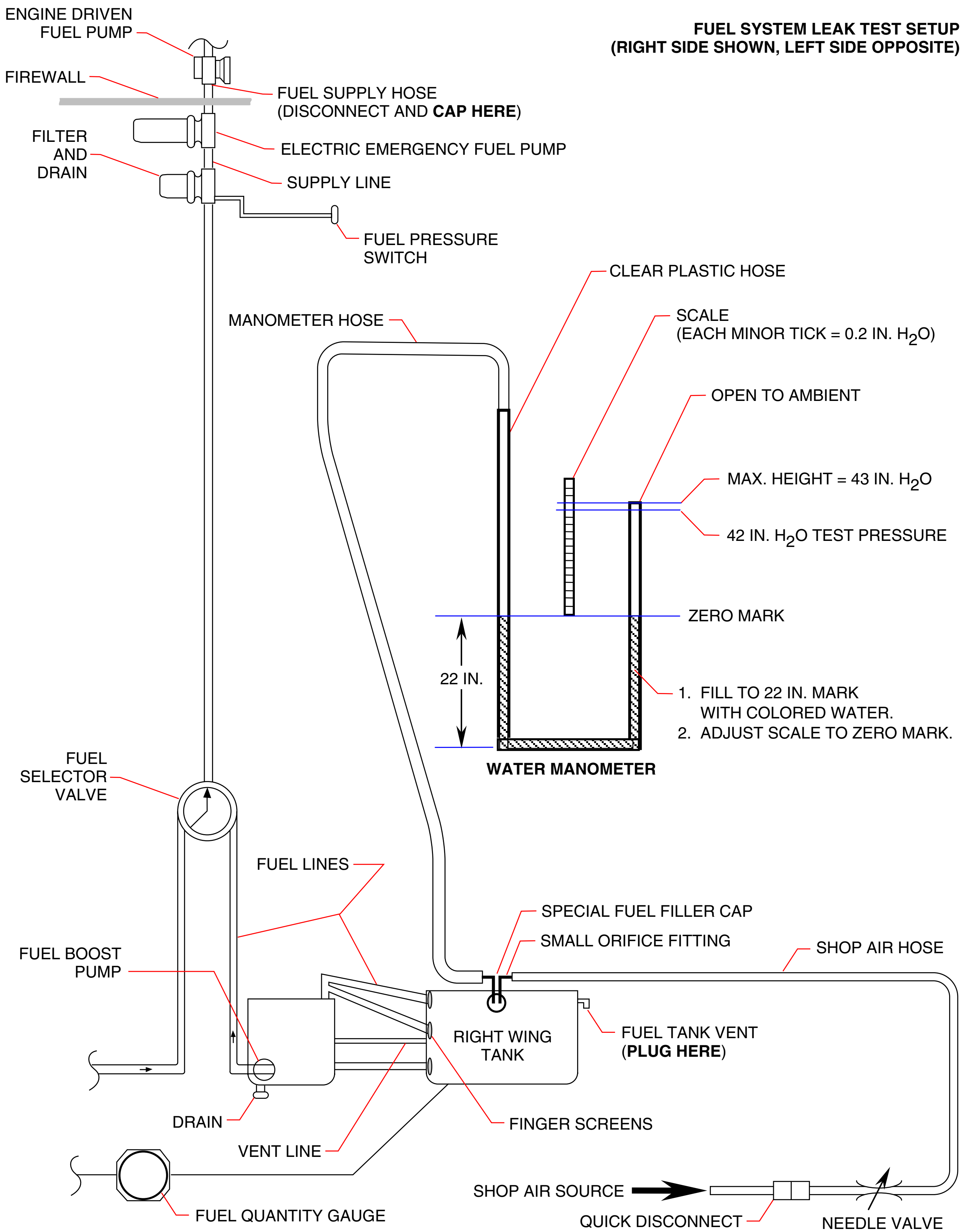
B. Fuel Plumbing.

WARNING: EVIDENCE OF FUEL LEAKAGE IN THE ENGINE COMPARTMENT MUST BE REPAIRED AND THE AREA CLEANED PRIOR TO STARTING ENGINE.

All such leaks must be repaired before further flight.



Categorizing Fuel Leaks in the Wet-Wing
Figure 2



Fuel System Leak Test Setup
Figure 3

3. Fuel System Leak Test

(PIR-PPS60049-2, Rev. G.)

A. Setup

(1) Locate the airplane appropriately and remove appropriate cowlings, access plates and panels to provide access to the fuel lines and fuel system components.

(2) Make available appropriate caps and plugs for closing lines and vents.

(3) Prepare a mild solution of soap and water for leak detection. Use only Colgate-Palmolive Co., Liquid M Mild 1 Hand Soap in this solution.

B. Test Fixture

(See “Figure 3” on page 28105.)

Construct or procure a test fixture with the following features:

(1) A quick-disconnect fitting for quickly removing shop air.

(2) A needle valve to shut-off shop air and hold pressure in the fuel system.

(3) A special fuel filler cap with fittings to accept shop air input (small orifice) and output to a water manometer.

NOTE: The shop air fitting must incorporate a small orifice (approximately 0.025 inch) to restrict shop air inflow and minimize the chance of over-pressurizing the system. Mark the small orifice fitting to identify it from the other fitting.

(4) Appropriate hoses to connect shop air and the manometer to the special fuel filler cap.

NOTE: To minimize the chance of cross-connecting shop air and the manometer to the wrong fitting, these hoses should be semi-permanently connected to the special fuel filler cap.

(5) The water manometer must be marked in inches at two-tenths of an inch intervals, have a height of no more than 43 inches, and must be open to the ambient air as a safety.

C. Procedure

NOTE: Possible areas for leaks in the system are all fuel system hoses, lines and component connections, wet-wing and collector tank access covers, fuel tank fillers and tank skin joints.

NOTE: Use appropriate caps for closing open lines and vents. Use soap and water mixture spray to check for leaks.

(1) Defuel the aircraft.

(2) Disconnect fuel supply hose from engine-driven fuel pump and cap hose.

(3) Check left wing system as follows:

(a) Install special fuel filler cap on left wing fuel filler.

NOTE: Verify that shop air hose is connected to the fitting with the small orifice and manometer hose is connected to the other fitting.

(b) Verify needle valve is fully closed and connect shop air to the quick disconnect fitting.

(c) Plug left wing fuel tank vent.

(d) Set fuel selector handle on LEFT tank position.

WARNING: IF MANOMETER FAILS TO RISE, CLOSE NEEDLE VALVE IMMEDIATELY AND DETERMINE CAUSE.

CAUTION: TO PREVENT FUEL TANK DAMAGE, DO NOT EXCEED 42 IN. H_20 (1.5 PSI).

(e) Slowly open needle valve and watch manometer for a rise in pressure.

(f) Allow manometer to rise to a test pressure of 42 ± ¼ In. H_2O and close needle valve.

(g) Disconnect shop air and let wing stabilize for a minimum of 20 minutes. Monitor leakage rate for thirty (30) minutes. A maximum leakage rate of 0.1 PSI within thirty (30) minutes is permissible. If the leakage rate exceeds this, detect the leak location and repair. Use soap solution as required.

(h) At the end of the thirty (30) minute period, turn the fuel selector valve handle rapidly to the RIGHT position, the OFF position, and back to the LEFT position. The right wing system should not pressurize during this test.

NOTE: Moving the selector valve handle rapidly reduces the pressure lost when the left and right ports overlap as the tanks are switched.

(i) Remove plug from left wing fuel tank vent and allow pressure to exhaust to ambient. Watch the manometer and verify that the vent relief valve closes between 27.7 In. H_2O (1.0 PSI) and 2.8 In. H_2O (0.1 PSI).

(j) Remove all caps, plugs, and test equipment. Proceed to right wing system check or reinstall fuel supply hose to engine-driven fuel pump, if finished.

(4) Check right wing system as follows:

(a) Install special fuel filler cap on right wing fuel filler.

NOTE: Verify that shop air hose is connected to the fitting with the small orifice and manometer hose is connected to the other fitting.

(b) Verify needle valve is fully closed and connect shop air to the quick disconnect fitting.

(c) Plug right wing fuel tank vent.

(d) Set fuel selector handle on RIGHT tank position.

WARNING: IF MANOMETER FAILS TO RISE, CLOSE NEEDLE VALVE IMMEDIATELY AND DETERMINE CAUSE.

CAUTION: TO PREVENT FUEL TANK DAMAGE, DO NOT EXCEED 42 IN. H_20 (1.5 PSI).

(e) Slowly open needle valve and watch manometer for a rise in pressure.

(f) Allow manometer to rise to a test pressure of 42 ± ¼ In. H_2O and close needle valve.

(g) Disconnect shop air and let wing stabilize for a minimum of 20 minutes. Monitor leakage rate for thirty (30) minutes. A maximum leakage rate of 0.1 PSI within thirty (30) minutes is permissible. If the leakage rate exceeds this, detect the leak location and repair. Use soap solution as required.

(h) At the end of the thirty (30) minute period, turn the fuel selector valve handle rapidly to the LEFT position, the OFF position, and back to the RIGHT position. The left wing system should not pressurize during this test.

NOTE: Moving the selector valve handle rapidly reduces the pressure lost when the left and right ports overlap as the tanks are switched.

(i) Remove plug from right wing fuel tank vent and allow pressure to exhaust to ambient. Watch the manometer and verify that the vent relief valve closes between 27.7 In. H_2O (1.0 PSI) and 2.8 In. H_2O (0.1 PSI).

(j) Remove all caps, plugs, and test equipment. Reinstall fuel supply hose to engine-driven fuel pump.

DISTRIBUTION

Fuel distribution consists of the collector tanks, two submerged electric boost pumps, fuel selector and switches, an inline fuel filter, an emergency electric fuel pump, and engine-driven fuel pump.

1. Collector Tanks

Each collector tank is accessible through the main landing gear wheel well access panel. The collector tank contains the electric boost pump, flapper valves and a drain valve.

NOTE: Drain wing tank and collector tank prior to removal of the collector tank access cover.

NOTE: Removal/installation sequence is the same for either left or right collector tank.

A. Removal

(1) Defuel the wing tank of the appropriate side.

(2) Ensure selector valve is in the OFF position.

(3) Drain remaining fuel from the collector tank.

(4) Remove the access panel from the collector tank and lower wing forward of the collector tank.

(5) Disconnect the electrical connection to the pump and the fuel line from the pump. Cap the open line to prevent any possible contamination.

(6) Remove the four nuts securing the collector tank and remove the tank.

B. Installation

(1) Install tank into place and secure with the four nuts.

(2) Remove the protective cap from the fuel line and connect to the fuel pump outlet.

(3) Connect the electrical connector to the pump leads.

(4) Ensure collector tank is clean, then install the tank access cover. Seal the access cover using MIL-S-8784.

(5) Reinstall the access panel on the lower wing.

2. Electric Boost Pumps

These pumps are located in each collector tank built into each wing wheel well area. Access to each pump is through the wheel well and removal of the sealed access cover on the side of the collector tank.

NOTE: Drain wing tank and collector tank prior to removal of the collector tank access cover.

A. Removal

(1) Remove collector tank per Collector Tanks, Removal, above.

(2) Remove the two screws securing the pump to the tank and remove the pump.

B. Installation

(1) Position fuel pump with gasket into the tank and secure in place with the two pump mounting screws.

(2) Install collector tank per Collector Tanks, Installation, above.

3. Fuel Drain Valve

A. Description

Fuel drain valves are located under each inboard wing. They are actuated either: by pushing up on the screwdriver slot for fuel check or by removal of the entire valve for rapid draining.

NOTE: Nut and nut retainer are secured inside tank and do not need to be removed.

B. O-ring Replacement

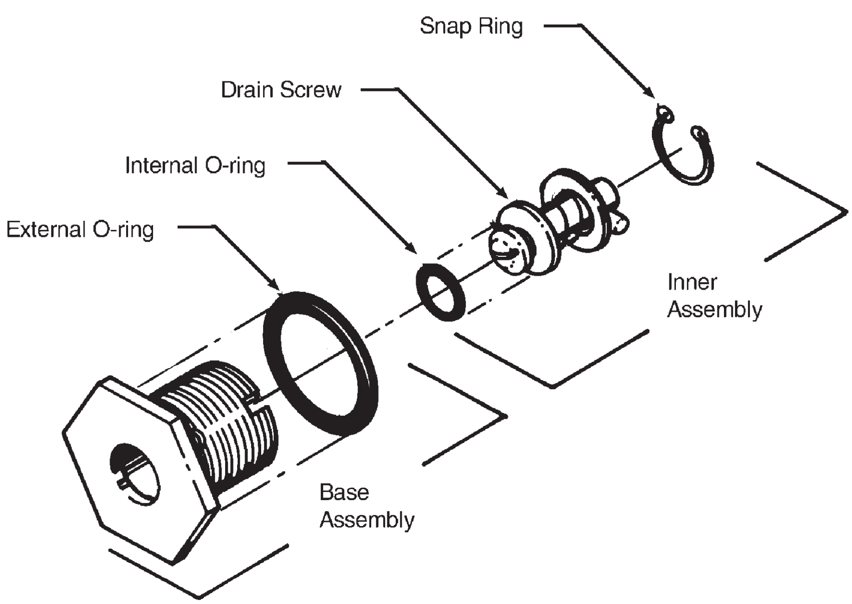
(1) Drain fuel by turning hexagonal base of drain valve. Remove drain valve from wing.
(2) Remove external O-ring located on hexagonal base.
(3) Remove snap ring to release inner assembly.
(4) Remove inner assembly by pushing up on drain screw with a screwdriver.
(5) Remove internal O-ring.
(6) Soak replacement O-rings in hydraulic fluid and coat with fluid-compatible gel.
(7) Install O-rings.
(8) Replace inner assembly. Replace snap ring.
(9) Install drain valve in wing.

4. Fuel Selector Valve

CAUTION: NO FIELD DISASSEMBLY OR REPAIR OF FUEL SELECTOR VALVES IS AUTHORIZED. MAINTENANCE IS LIMITED TO REMOVAL AND REPLACEMENT OF THE WHOLE UNIT.

A. Description

The selector valve is located behind an access panel (pressure closeout plate) on the right hand side of the fuselage interior. The valve is mounted aft of the F.S. 139.8 frame in a compartment that is sealed off from the pressurized interior.



Fuel Drain Valve
Figure 1

B. Removal

(1) Drain both fuel tanks. (Refer to 12-10-00, Draining Fuel System.)

(2) Gain access to the pressure closeout panels on the right hand side of the fuselage interior.

(3) Loosen the hose clamps which secure the fuel line pressure seal halves to the fuel line going to the left tank.

(4) Remove the screws and washers that secure the base of the pressure closeout plate to the frames. Slide the pressure closeout plate inboard along the fuel line going to the left tank.

(5) Disconnect the fuel selector cable from the actuating arm on the bottom of the fuel selector valve.

(6) Disconnect the switches electrical connector.

(7) At the fittings on the valve, remove the three fuel lines that connect to the valve.

(8) Remove the screw that holds the fuel selector support to the frame.

C. Installation

(1) Position the fuel selector valve in the compartment. Secure the valve support to the frame with the screw.

(2) Attach the three fuel lines to the valve.

(3) Reconnect the switches' electrical connector.

(4) Reconnect the fuel selector cable to the actuating arm on the bottom of the fuel selector valve.

(5) Secure the base of the pressure closeout plate to the frames with screws and washers.

(6) Tighten the hose clamps on the fuel line pressure seal halves which are on the fuel line going to the left tank.

(7) Fill the fuel tanks and check for proper operation of the fuel selector valve.

D. Switches Adjustment

(See "Figure 2".)

With the selector valve/switch removed from the airplane:

(1) Ensure that the switch has a pin overtravel of 0.06 in.

(2) When the handle is moved to its various positions, ensure that:

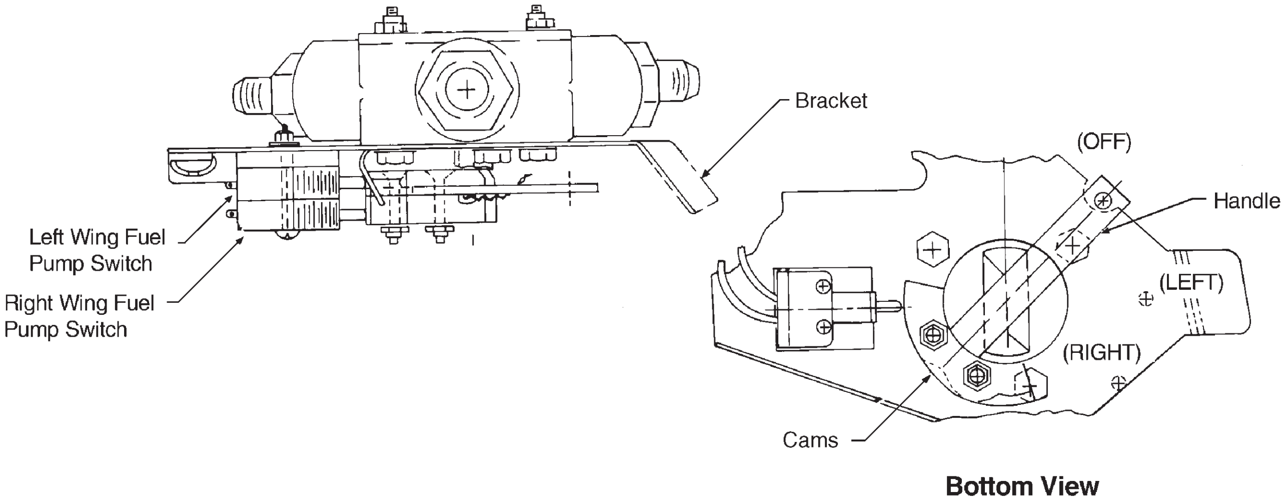
(a) The left switch should make contact (closed circuit) when the handle is in its mid position. The left switch should be an open circuit when the handle is in the off and right positions.

(b) The right switch should make contact (closed circuit) when the handle is in the right position. The right switch should be an open circuit when the handle is in the off and left positions.

(3) Ensure that the plungers point to the axis of handle rotation. Adjust the switches on the cam by moving them towards or away from the cam, not by twisting the switches to one side.

5. Fuel Selector Cable Adjustment

Adjust fuel selector cable so that the fuel selector arm will be in the (center) detent position when the selector knob, located on the instrument panel, is moved to the left tank (center) position from the right tank position. Visually check the fuel selector arm and cam to confirm proper adjustment. Properly adjusted, the selector knob will not reach the instrument panel slot travel limits when resting in the right tank and off positions. Travel should be equal distance from center ± .10.



Fuel Selector Switch and Valve
Figure 2

6. Fuel Filter Assembly

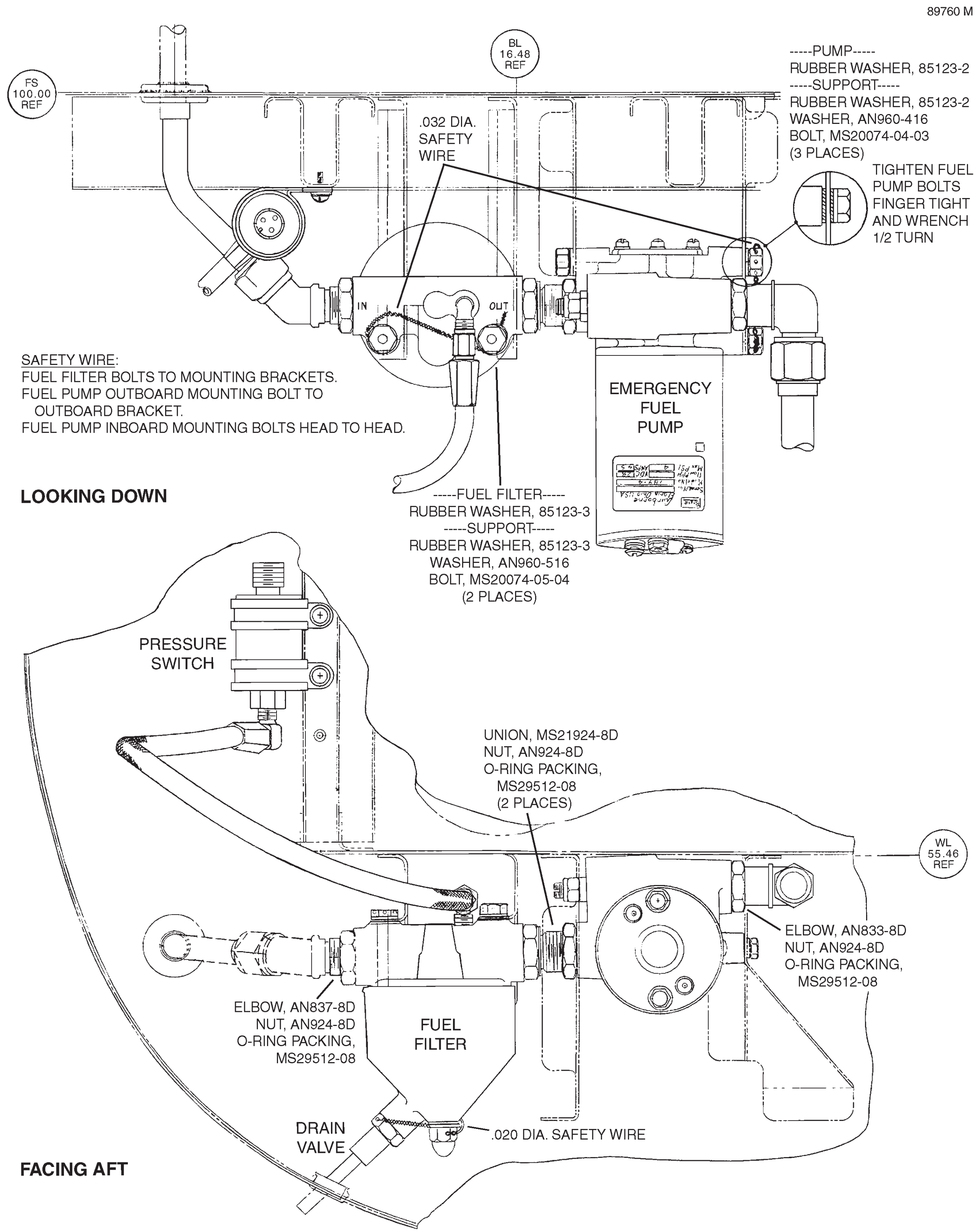
(See Figure 3.)

Access to the fuel filter assembly is through the forward baggage compartment, right side floor panel access cover. The fuel filter assembly is mounted on brackets attached to the forward, right hand side of the forward pressure bulkhead below the baggage compartment floor.

NOTE: For routine filter servicing, refer to 12-10-00, Fuel System.

A. Removal

(1) Ensure electrical power is OFF.
(2) Ensure fuel selector control is in the OFF position.
(3) Drain all fuel from engine to shutoff valve through outside fuel filter drain valve.
(4) Remove the forward baggage compartment carpet.
(5) Remove the right side floor panel access cover.
(6) Disconnect electrical leads from emergency fuel pump.
(7) Disconnect and cap fuel supply hose from emergency fuel pump fitting.
(8) Disconnect and cap fuel supply hose from fuel filter inlet.
(9) Disconnect pressure switch fuel hose from fuel filter.
(10) Remove safety wire and two mounting bolts from fuel filter.
(11) Remove safety wire and three mounting bolts from emergency fuel pump.
(12) Remove emergency fuel pump and filter assembly from aircraft.
(13) Loosen jam nut at fuel pump inlet fitting.
(14) Remove fuel filter assembly.



Fuel Filter and Emergency Fuel Pump
Figure 3

B. Installation

NOTE: Be sure to replace O-rings at inlet and discharge pump fittings.

(1) Position emergency fuel pump onto fuel filter and tighten jam nut at emergency fuel pump inlet fitting.

(2) Position emergency fuel pump and fuel filter assembly into place in aircraft with the filter inlet port facing the outboard skin and the drain valve inserted in the lower opening of the fuselage.

(3) Secure emergency fuel pump to mounting brackets with three mounting bolts, washer sets and safety wire.

(4) Secure the filter assembly to its mounting brackets with two mounting bolts, washer sets and safety wire.

(5) Connect pressure switch fuel hose to fuel filter.

(6) Remove protective cap and connect fuel supply hose to fuel filter inlet.

(7) Remove protective cap and connect fuel supply hose to emergency fuel pump fitting.

(8) Connect electrical leads to emergency fuel pump.

(9) Position fuel selector control to one of the ON positions and check fittings for leaks. Upon completion of check return selector to the OFF position.

(10) Install the floor panel access cover and secure in place.

(11) Install the forward baggage compartment carpet.

7. Emergency Fuel Pump

(See “Figure 3” on page 28205.)

Access to the emergency fuel pump is through the forward baggage compartment, right side floor panel access cover. The emergency fuel pump is mounted on brackets attached to the forward, right hand side of the forward pressure bulkhead below the baggage compartment floor.

A. Removal

See Fuel Filter Assembly “Removal” on page 28204.

B. Installation

NOTE: Be sure to replace O-rings at inlet and discharge pump fittings.

See Fuel Filter Assembly “Installation”.

CHART 1
ELECTRIC FUEL PUMP SPECIFICATIONS

DESCRIPTION	AIRBORNE 2C6-7	AIRBORNE 1B9-4 EMERGENCY
Voltage (Vdc) Current (Amps)	28 4.0 (Max.)	28 6.0 (Max.)
Fuel Flow Fuel Pressure	0 - 340 PPH 7 (MIN.) - 12 (MAX.)	0 - 375 PPH 35 (MIN.) - 47 (MAX.)
Proof pressure (Zero external leakage)	60 PSI	60 PSI

INDICATING

NOTE: S/N's 4636021 and up incorporate Integrated Engine Instrumentation Systems. See 77-40-00 for more information.

1. Fuel Quantity Senders

(PIR-88535, Rev. B.)

NOTE: This section incorporates requirements from Piper Service Letter No. 1159B.

In PA-46-350P S/N's 4636001–4636651 less 4636633, and all PA-46R-350T airplanes; fuel senders are located within the wing tank at W.S. 125.00 and 163.15 as seen in "Figure 1". These units are mounted on a bracket which are part of the forward access panel in the lower surface of the wing tank. These senders transmit electrically the quantity of the fuel in each tank to the fuel quantity gauge/indicator/display mounted in the instrument panel.

CAUTION: THE ORIGINAL WIRE-WOUND FUEL QUANTITY SENDERS ARE NO LONGER AVAILABLE AND NEW THICK-FILM FUEL QUANTITY SENDERS HAVE BEEN INTRODUCED. FAILURE OR REPLACEMENT OF ONE WIRE-WOUND FUEL QUANTITY SENDER REQUIRES THE REPLACEMENT OF ALL FOUR EXISTING FUEL QUANTITY SENDERS (TWO IN EACH WING) WITH THE CORRECT FUEL SENDER REPLACEMENT KIT AS FOLLOWS:

GROUP A		
AIRCRAFT MODEL	SERIAL NUMBERS	PIPER KIT P/N
PA-46-350P	4636001 thru 4636195	88535-001
	4636196 thru 4636379	88535-002
	4636380 thru 4636462	88535-003
PA-46R-350T	4692001 thru 4692133	88535-004

THE NEW THICK-FILM FUEL QUANTITY SENDERS WERE FACTORY INSTALLED IN THE FOLLOWING AIRPLANES:

GROUP B	
AIRCRAFT MODEL	SERIAL NUMBERS
PA-46-350P	4636463 THRU 4636632 4636634 THRU 4636651
PA-46R-350T	4692134 THRU 4692214

FOR SERVICE REPLACEMENTS PARTS FOR THE FUEL SENDER ONLY, BRACKET ONLY, OR COMPLETE FUEL SENDER ASSEMBLY, SEE "CHART 1".

NOTE: In airplanes equipped with Avidyne Entegra, the new thick-film fuel quantity senders will perform optimally when the DAU software is upgraded to Revision G or later. This requires exchanging your existing DAU for an upgraded unit.

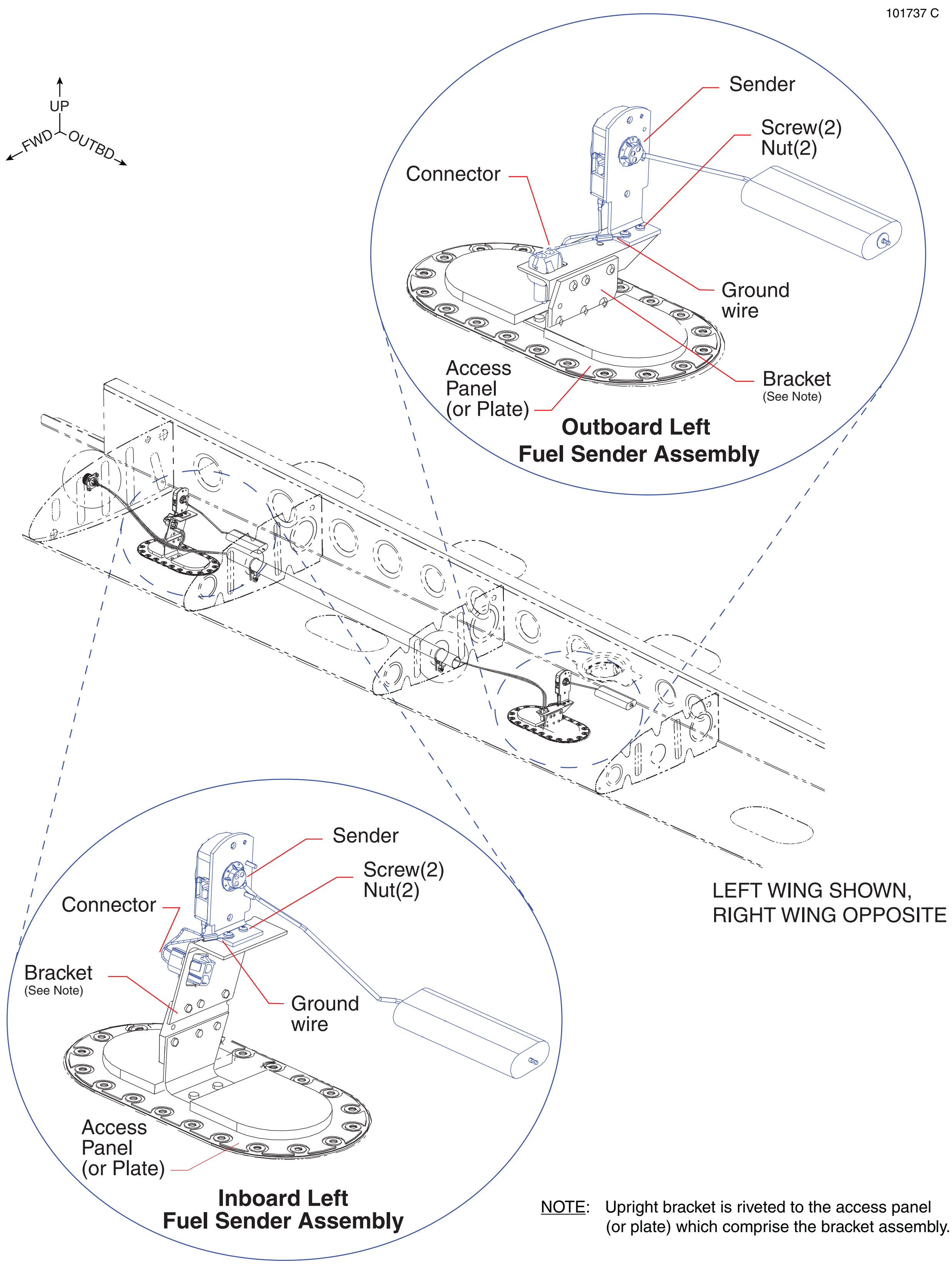
NOTE: If a wing or any section of a wing has been replaced while in service, fill out the Wing Replacement Form provided in Piper Service Letter No. 1159B and email/fax it to Piper to determine the appropriate replacement kit.

CHART 1
FUEL SENDER REPLACEMENT PARTS

FUEL SENDER LOCATION	FUEL SENDER ONLY P/N(1)	COMPLETE FUEL SENDER ASSEMBLY P/N (Fuel Sender Assembly and Bracket Assembly)	
		GROUP A(2)	GROUP B(2)
Inboard Left	601-649 or 601-653	106755-009 or 106755-809	106755-008 or 106755-808
Outboard Left	601-650 or 601-654	106756-009 or 106756-809	106756-008 or 106756-808
Inboard Right	601-651 or 601-655	106757-009 or 106757-809	106757-008 or 106757-808
Outboard Right	601-652 or 601-656	106758-009 or 106758-809	106758-008 or 106758-808

(1) For each location, all part numbers listed are for the latest configuration fuel senders in that location. Order these replacements for the fuel senders only.
(2) Group A and Group B refer to the effectivity charts of the same name from the CAUTION within “Fuel Quantity Senders” on page 28401.

AIRCRAFT MODEL	SERIAL NUMBER	BRACKET ONLY P/N	
		INBOARD BRACKET (Left of Right)	OUTBOARD BRACKET (Left of Right)
PA-46-350P	4622001 thru 4636379	106753-003	106754-003
	4636634 thru 4636651 4636380 thru 4636632	106753-002	106754-002
PA-46R-350T	4692001 thru 4692214	106753-002	106754-002



NOTE: Upright bracket is riveted to the access panel (or plate) which comprise the bracket assembly.

Fuel Sender Installation (Typical)
Figure 1

A. Removal

CAUTION: DO NOT BEND THE SENDER FLOAT ROD WHEN HANDLING THE FUEL SENDER.

(1) Drain all fuel from the wing tank being serviced.

(2) Set the battery/master switch to the OFF position and ensure no external power is connected.

(3) Using a temporary marker or tape, mark the sender access panel to indicate aircraft forward.

(4) Remove the sender access cover as follows:

(a) In S/N's 4636001 thru 4636379, remove the screws securing the access cover. With a suitable tool, pry the access cover away from the wing, taking care not to damage the mating surfaces - especially on the wing. When the access cover becomes free, carefully lower the cover and sender out of the wing until the electrical connection can be reached and disconnected.

(b) In S/N's 4636380 and up and S/N's 4692001 and up, remove the screws securing the access cover and carefully lower the cover and sender out of the wing until the electrical connection can be reached and disconnected.

(5) Note the position of the fuel sender's orientation, location, and installation prior to removing the sender from the bracket.

B. Installation

CAUTION: DO NOT BEND THE SENDER FLOAT ROD WHEN HANDLING THE FUEL SENDER.

NOTE: In S/N's 4636380 and up and PA-46R-350Ts, whenever fuel quantity senders are replaced, perform the Fuel Quantity Indicator Calibration (S/N's 4636375 and up) (S/N's 4692001 and up), below.

NOTE: In S/N's 4636001 thru 4636379, if any press fit / clinch nut in a sender unit/access cover retainer becomes loose and spins, it may be replaced with a riveted nutplate as follows:

Press Fit / Clinch Nut	replaced with	Nutplate	and	Rivets
PS10062-10-2 (467-511)		MS21059L08 (477-809)		MS20426AD3
PS10062-10-3 (603-312)		NAS1474A08 (477-899)		MS20426AD3

(1) Install sender unit on its bracket (if previously removed). Ensure the sender unit is properly positioned.

(2) Connect the electrical leads (from inside the tank) to the sender assembly.

CAUTION: ENSURE THAT WIRES ARE ROUTED ON FORWARD SIDE OF SENDER PRIOR TO SECURING SENDER PLATE TO WING.

(3) In S/N's 4636001 thru 4636379, ensure that the sender unit/access cover is properly positioned in relation to the wing, apply Products Research Co. (PRC) PR1428-B-2 (or PR1403G-B-2) sealant and install the sender unit/access plate on the wing with screws. Allow sealant to cure 72 hours before refilling tanks.

(4) In S/N's 4636380 and up and S/N's 4692001 and up, ensure that the sender unit/access cover is properly positioned in relation to the wing, and install the sender unit/access plate with screws. Torque the screws per Figure 1, 28-10-00.

2. Fuel Quantity Probes

Refer to “Figure 2” on page 28406. In S/N’s 4636633, 4636652 thru 4636775; fuel probes are installed in each wing at W.S. 116.89, 167.59, and 207.23. In S/N’s 4636776 and up, fuel probes are installed in each wing at W.S. 122.99, 167.59, and 207.23. In each case, the fuel probe clips into a bracket mounted on a rib on the forward side of the main spar.

CAUTION: WHILE THE WIRING, INSTALLATION, AND REMOVAL OF THE PROBES IS THE SAME, EACH PROBE IS A DIFFERENT LENGTH UNIQUE TO ITS PARTICULAR WING STATION.

CAUTION: RECALIBRATE THE FUEL QUANTITY INDICATION SYSTEM PER “FUEL QUANTITY INDICATOR CALIBRATION (GARMIN G1000)” ON PAGE 284018 OR “FUEL QUANTITY INDICATOR CALIBRATION (GARMIN G1000 NXI)” ON PAGE 284020 WHENEVER A FUEL PROBE IS REPLACED.

A. Troubleshooting

If fuel quantity is displayed as red X (G1000) or yellow X (G1000 NXi), the following procedure may be used to assist in determining potential fuel probe failures. See also Troubleshooting information in 34-25-01 (G1000) or 34-25-02 (G1000 NXi) for the PFD and the No. 1 GIA 63W.

(1) Ensure all electrical buses are powered, electrical and avionics equipment is powered.

(2) With MFD on splash screen, select soft keys 1, 2, 3, 1 in sequence. 1 is leftmost key.

(3) Observe CAS message list for any Fuel Quantity Invalid messages.

NOTE: The number displayed in the message represents the invalid fuel probe, as sensed by the fuel quantity processor. The probes are numbered in sequence as shown in Identification, above.

B. Removal

(1) Turn master switch OFF.

(2) Defuel appropriate wing. (See 12-10-00, Draining Fuel System.)

(3) Remove the appropriate access plate(s) on the bottom of the wing for the fuel probe(s) to be removed.

(4) Reaching inside the wing, locate the fuel probe and pull it from the spring clip(s).

(5) Pull the fuel probe to the access hole and disconnect the electrical lead.

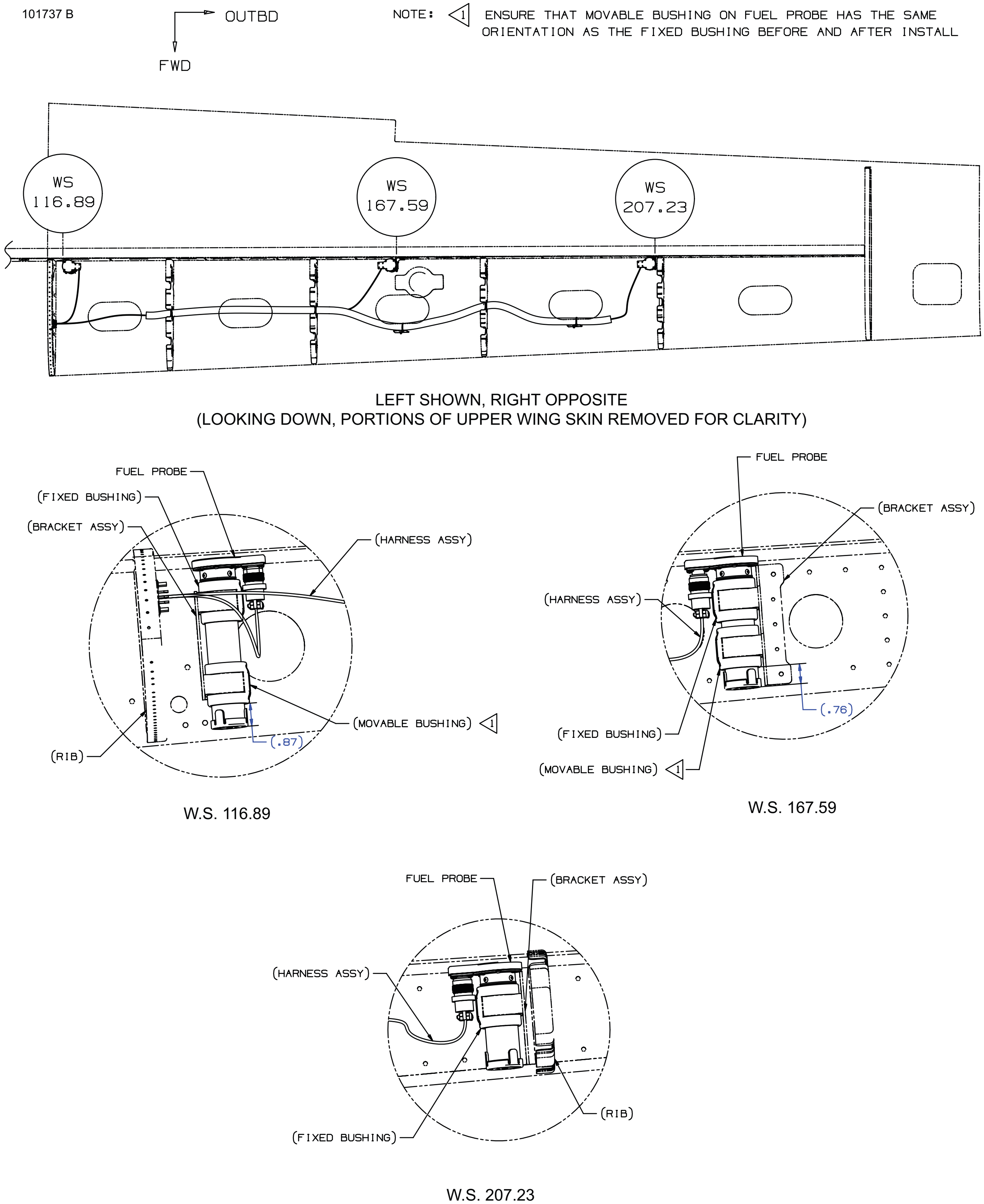
C. Installation

(1) Connect the electrical lead to the fuel probe.

(2) Reaching inside the wing, push the probe into its spring clip(s). Take care to ensure the anti-rotation tab on the upper spring clip engages the slot in the collar on the probe.

NOTE: Ensure the bottom of each installed probe clears the aircraft structure and/or fuel tank sealant by a minimum of 0.125 inch.

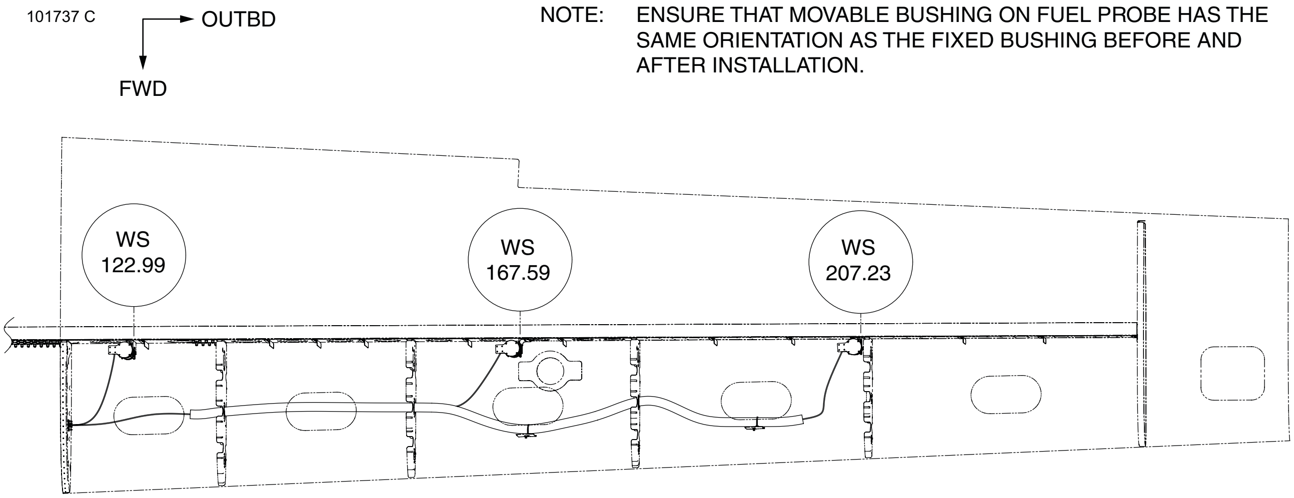
(3) Replace access cover(s) and install per Access Covers, Installation, 28-10-00.



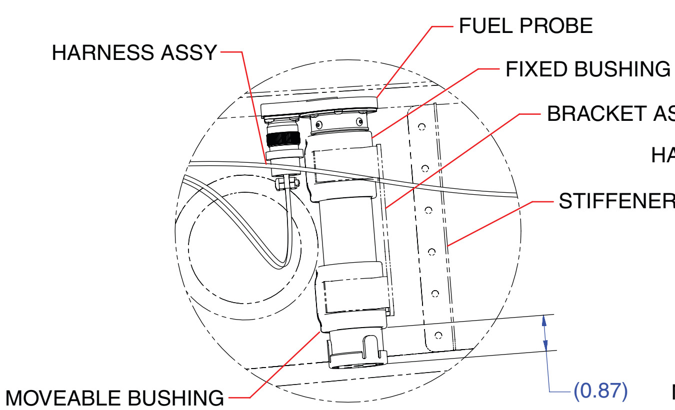
Effectivity
4636633, 4636652–4636775

Fuel Quantity Probes
Figure 2 (Sheet 1 of 2)

NOTE: ENSURE THAT MOVABLE BUSHING ON FUEL PROBE HAS THE SAME ORIENTATION AS THE FIXED BUSHING BEFORE AND AFTER INSTALLATION.



LEFT SHOWN, RIGHT OPPOSITE
(LOOKING DOWN, PORTIONS OF UPPER WING SKIN REMOVED FOR CLARITY)



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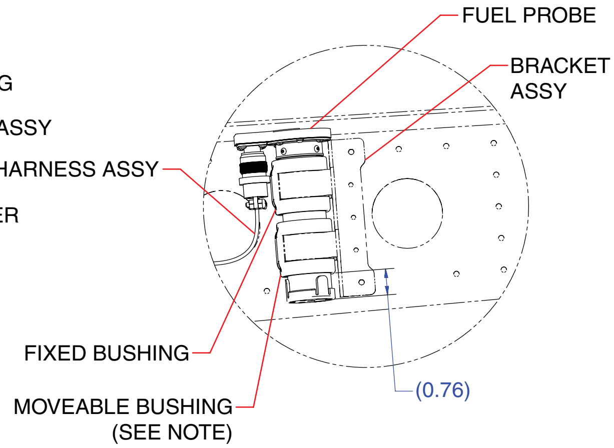

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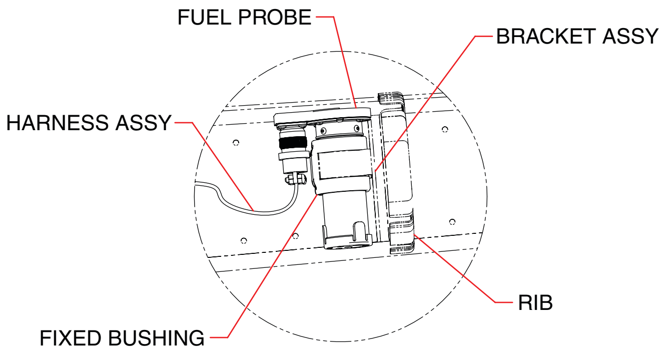


W.S. 207.23

Fuel Quantity Probes
Figure 2 (Sheet 2 of 2)

Effectivity
4636776 and up

3. Fuel Quantity Gauge Accuracy Test (S/N's 4636001 thru 4636020 only.)

(PIR-PPS60032-8, Rev. B.)

A. Pre-Installation Check.

Prior to installation, bench test the gauge as follows:

(1) Install dummy resistor across left or right sender as shown in “Figure 3”.

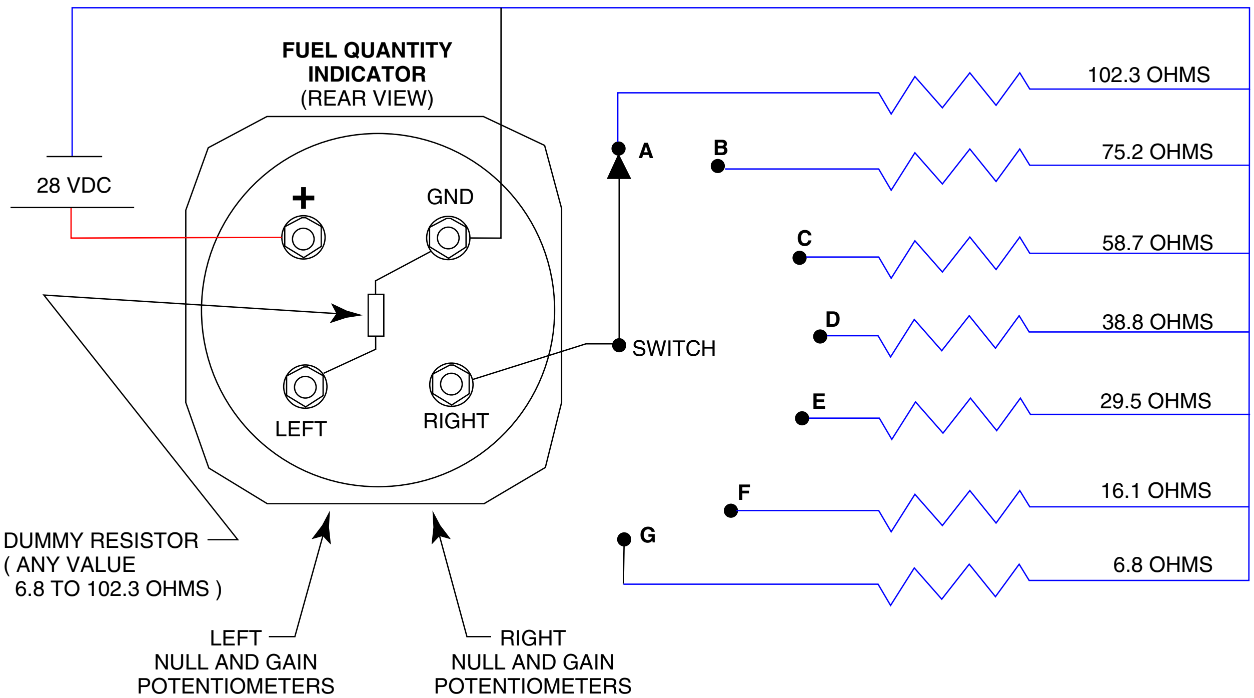
(2) Connect resistance decade across sender terminals not used by dummy resistor.

(3) Apply 28 ±1 VDC to the positive and negative terminals at the rear of the gauge and through the resistance decade as shown in “Figure 3”.

(4) Low end adjustment: Select position “G” on resistor decade. Verify that instrument needle points to “E.” If not, adjust respective “NULL” potentiometer to center needle on “E” radial.

(5) High end adjustment: Select position “A” on resistor decade. Verify that instrument needle points to “F.” If not, adjust respective “GAIN” potentiometer to center needle on “F” radial.

(6) Full range check: After low and high end adjustments have been made, verify that for each resistive value the gauge indication is as specified in “Chart 1”.



Pre-Installation Test Set-up
Figure 3

Effectivity
4636001–4636020

B. Post-Installation Functional Check.

(1) Set up:

(a) Level the airplane ±1 degree laterally and longitudinally. See 8-20-00.

(b) Place the battery switch in the OFF position.

(c) Connect the external power supply unit.

(d) Adjust the power supply to provide 28 ±1 VDC.

(e) Continue with procedure below, record test results when completed.

(2) Low end gauge check: With the tanks completely dry, turn the fuel selector to Left or Right. Add one gallon (unusable) fuel to each tank. Check that the needle of each gauge points to “E.” If not, adjust the respective “NULL” potentiometer until the needle centers on “E” (± 0, -1/2).

(3) High end gauge check: With the tanks full, check that the needle of each gauge points to “F.” If not, adjust the respective “GAIN” trim potentiometer until the needle centers on “F.”

(4) Full range gauge check, use either of the following methods:

(a) With the tanks full, defuel each tank fifteen gallons at a time until 15 gallons is reached and then, five gallons at a time until empty. Check the gauge indication at each increment against the tolerances specified in “Chart 2”.

(b) Starting with the tanks empty, add fuel to each tank to the increments specified in “Chart 2”. Check the gauge indication at each stage against the tolerances specified in “Chart 2”.

NOTE: With either of the above methods, be sure and vibrate the tank by hand bumping its lower surface and vibrate the gauge by finger tapping the glass at each measurement point.

CHART 1
PRE-INSTALLATION GAUGE CHECK

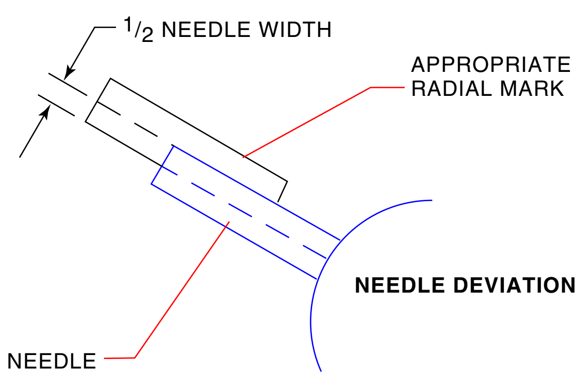
SWITCH* POSITION	RESISTANCE IN OHMS	GAUGE INDICATION	TOLERANCE** (NEEDLE WIDTHS)
G	6.8	E	+ 0, - 1/2
F	16.1	5	± 1/2
E	29.5	10	± 1/2
D	38.8	15	± 1/2
C	58.7	30	± 1/2
B	75.2	45	± 1/2
A	102.3	F	± 1/2

* - SEE “Figure 3”
** - SEE “Figure 4”

CHART 2
FULL RANGE CHECK

TOTAL FUEL QUANTITY (US GAL) (INCLUDES 1 GAL UNUSABLE)	GAUGE INDICATION	TOLERANCE* (NEEDLE WIDTHS)
1	E	+ 0, - 1
6	5	± 3/4
11	10	± 3/4
16	15	± 1
31	30	± 1 1/2
46	45	± 1 1/2
60	F	± 1 1/2

* - SEE “Figure 4”.



Needle Deviation
Figure 4

4. Fuel Quantity Gauge Calibration (S/N's 4636021 thru 4636374.)

(PIR-PPS60032-13, Rev. Q.)

NOTE: S/N's 4636021 thru 4636374 - Review Chapter/Section 77-40-00: Transicoil Electronic Module Instrument System (EMIS) to ensure a basic understanding of the system components and their operation before proceeding

NOTE: S/N's 4636021 thru 4636131 only - If utilizing resistance measurements from "Chart 4", it will be necessary to connect a digital ohmmeter per "Chart 4". To troubleshoot outputs from fuel tank senders, disconnect harness from back of gauge and read resistance across the appropriate pins.

A. Setup

(1) Level the airplane ±1 degree laterally and longitudinally. See 8-20-00.

(2) Place the battery switch in the OFF position.

(3) Connect the external power supply unit.

(4) Adjust the power supply to provide 28 ±1 VDC.

(5) Continue with procedure below, record test results in "Chart 4" when completed.

B. Non-Programmable Transicoil EMIS Gauge (S/N's 4636021 thru 4636131 only)

NOTE: The Programmable Transicoil EMIS Gauge (P/N 602-222) installed in S/N's 4636132 thru 4636374 as original equipment is backwardly compatible and may be found installed in S/N's 4636021 thru 4636131 as a service spare. These modules can be identified by the absence of potentiometers and the presence of a calibration switch on the left side of the module. If found, calibrate per Programmable Transicoil EMIS Gauge, below, substituting Quad Digital Indicator (QDI) for each reference to the Enhanced Digital Indicator (EDI).

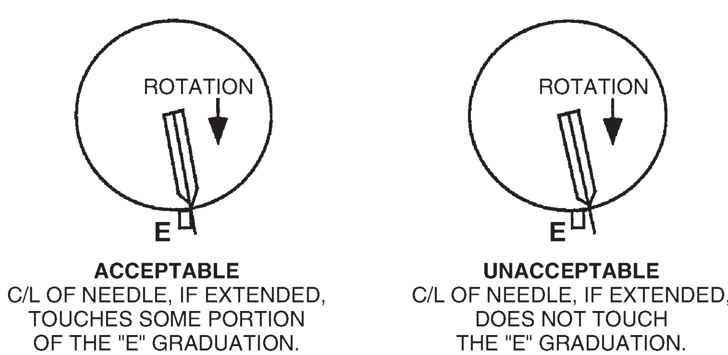
(1) Low End Gauge Check

CAUTION: USE NON-CONDUCTIVE BLADE SCREWDRIVER TO ADJUST POTENTIOMETERS.

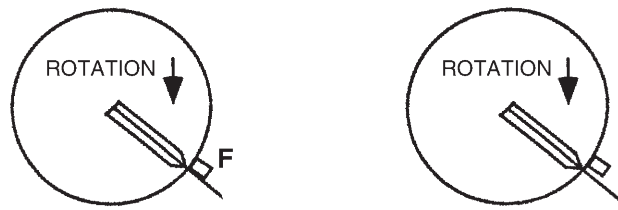
With both tanks completely dry, add one gallon (unusable) fuel to each tank. After the measured amount has been put in the tank, vibrate the tank by bumping its lower surface. Adjust potentiometers marked "Z" (clockwise to lower reading) until the analog pointers are at the center of the "E" (empty) graduations. Allow indicators to stabilize for 30-40 seconds. Readjust as required. Check digital readout for a "0" gallon indication. If digital does not indicate "0", readjust appropriate "Z" potentiometer until both analog and digital read zero.

NOTE: The potentiometers are in cavities on the right and left sides of the fuel quantity module (right potentiometer adjusts right gauge, left adjusts left gauge).

NOTE: Needles on analog gauges must not go below the "E" (empty) graduations. See"Figure 5". Tolerance on ''E" (empty indication) is ±0 gallons.



"E" (Empty) Needle Tolerance
Figure 5



ACCEPTABLE
C/L OF NEEDLE, IF EXTENDED, TOUCHES SOME PORTION OF THE "F" GRADUATION.

UNACCEPTABLE
C/L OF NEEDLE, IF EXTENDED, DOES NOT TOUCH THE "F" GRADUATION.

"F" (Full) Needle Tolerance
Figure 6

(2) Full End Gauge Check

Add 60 gallons of fuel to each tank. After the measured amounts have been put in tanks, vibrate the tanks by bumping their lower surfaces. Adjust "S" (span) potentiometers (counter-clockwise to lower reading) so the analog needles point to "F" (full) graduations. Turn "S" potentiometers so digital indicators read 58 gals. Turn "S" potentiometers in opposite direction until digital readings just change from 59 to 60 gals. At this point turn the "S" potentiometers 1/4 turn more in the increasing reading direction. Permissible to have analog needles touching top edges of (F) graduations after the adjustments have been made. Allow indicators to stabilize for 30-40 seconds. Readjust as required. Check digital readout for 60 gallon indication. If digital does not indicate "60", readjust "S" (span) potentiometer so both analogs and digital read "60".

NOTE: The potentiometers are in cavities on the right and left sides of the fuel quantity module (right potentiometer adjusts right gauge, left adjusts left gauge).

NOTE: Needles on analog gauges must not go above "F" (full) graduations. See "Figure 6".

C. Programmable Transicoil EMIS Gauge (S/N's 4636132 thru 4636374)

(1) Initiation of Calibration Mode.

The left and right Fuel Quantity Module Analog Indicators are calibrated individually. The following instructions specifically address the left indicator.

(a) If fuel quantity indicator is installed in instrument panel, remove gauge from panel far enough to move calibration switch on left side of the module from "RUN" to "CAL" position (move switch toward front of gauge). This will place left fuel quantity instrument in auto-calibration mode.

NOTE: Instrument power must be maintained to the Enhanced Digital Indicator (EDI) and Fuel Quantity Module during the entire process. Any interruption of power to the Fuel Quantity Module will reset the calibration sequence to 0.

(b) When the calibration switch is moved to "CAL" position, the EDI is automatically selected and a fuel quantity will be displayed in the left window. Also, the yellow LED for the tank being calibrated will be flashing.

(c) If the sensor input resistance from the left tank is within tolerance for the first calibration step (E-0) as specified in "Chart 3", then the green and yellow LED's will flash alternately. If the left sensor is out-of-range, only the yellow LED will flash on the left indicator. See "Figure 8".

CHART 3
FUEL QUANTITY MODULE DUAL ANALOG INDICATOR CALIBRATION RESISTANCE

Calibration Points - GAL	E-0	5	15	30	45	F-60
Minimum Ohms	1.75	13.6	32.15	52.55	72.85	98.75
Nominal Ohms	6.75	18.6	42.15	62.55	80.85	108.75
Maximum Ohms	11.75	28.6	51.15	70.55	90.85	113.75
Acceptable Tolerance	±5	-5/+10	-10/+9	-10/+8	-8/+10	-10/+5

(2) Calibration Procedure

NOTE: The select button (see “Figure 7”) on the front of the fuel quantity module analog indicator being calibrated will be used to set the calibration points and advance to the next step.

(a) Add one (1) gallon (unusable) fuel to the left tank and allow the instrument and fuel to settle for two (2) minutes.

NOTE: At each calibration step, vibrate the tank by bumping its lower surface.

(b) After two (2) minutes, the green and yellow LED’s on the module indicator will be flashing alternately, the EDI will display “0” and the analog gauge will display “E” (see “Figure 6”).

(c) Press the left “SEL” button and hold for two (2) seconds. If the sensor is in tolerance for the step being recorded, the LED’s will stop flashing and remain on, the external alarm will sound and the EDI display and the fuel quantity module pointer will advance to the next step. The external alarm will continue to sound as long as the “SEL” button is held.

NOTE: If the sensor is out-of-range for the calibration step being programmed, then pressing the left “SEL” button will not advance the module to the next step.

(d) The calibration step sequence is: “E”, 5, 15, 30, 45, and “F”.

(e) When the module advances to the next step, normally only the yellow LED will flash until fuel is added.

(f) Add the appropriate amount of fuel for the fuel point displayed. Do not press the left “SEL” button when the green LED starts to flash. Allow the fuel and module to settle for two (2) minutes, then press the left “SEL” button and advance to the next step.

(g) When 60 gallons of fuel have been added and a two (2) minute settling time has passed, press the left “SEL” button. When you release the button, only the green LED will be flashing. This indicates that the calibration is finished.

(h) Reset the left indicator calibration switch (on the left side of the fuel quantity module) from “CAL” to “RUN” (i.e. - move it towards the rear of the module). This will return the module to normal operating mode.

(i) Initiate calibration for the right fuel quantity module analog indicator per (1) above and then steps (2) (a) thru (h). The right indicator calibration switch is on the right side of the fuel quantity module.

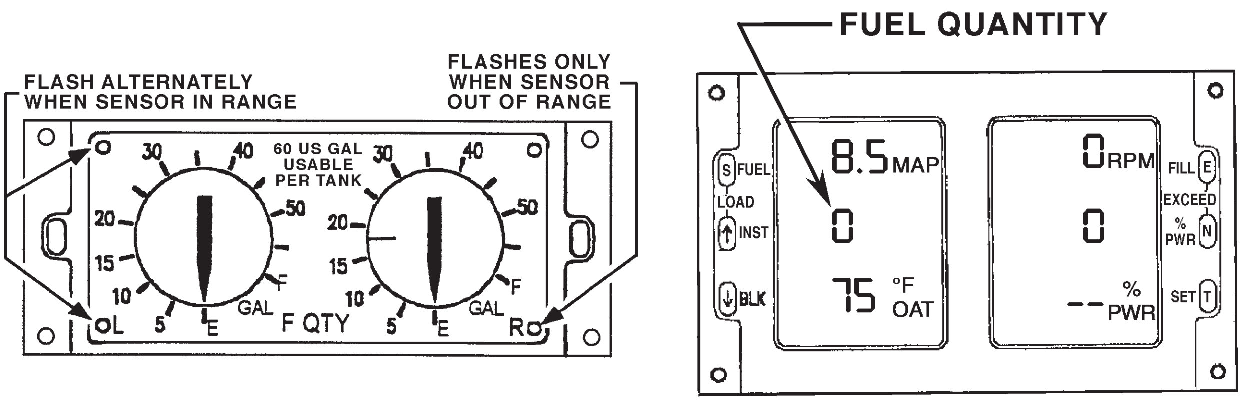
(j) Reinstall the fuel quantity module in the instrument panel.

D. Complete procedure as follows:

(1) If used, disconnect the digital ohmmeter.

(2) Disconnect the external power supply unit.

(3) Replace access covers.



(Both shown at "E-0" Calibration Step.)

Fuel Quantity Module Dual Analog Indicator — Enhanced Digital Indicator (EDI)

Figure 7

5. Fuel Gauge Full Range Check (S/N's 4636021 thru 4636374.)

(PIR-PPS60032-13, Rev. Q.)

NOTE: Full Range Check for S/N's 4636001 thru 4636020 only, is provided under Fuel Quantity Gauge Accuracy Test, above.

A. Setup

See Fuel Quantity Gauge Calibration, Set up, above.

B. Non-Programmable Transicoil EMIS Gauge (S/N's 4636021 thru 4636131 only).

NOTE: The Programmable Transicoil EMIS Gauge (P/N 602-222) installed in S/N's 4636132 thru 4636374 as original equipment is backwardly compatible and may be found installed in S/N's 4636021 thru 4636131 as a service spare. These modules can be identified by the absence of potentiometers and the presence of a calibration switch on the left side of the module. If found, full range check per Programmable Transicoil EMIS Gauge, below.

(1) Defuel both fuel tanks. As fuel level decreases to 5 +0/-1 gals, verify the low fuel alarms sound.

(2) When defueling is complete, verify that the needles on analog gauges are pointing to the areas of the “E” graduations as described in Fuel Quantity Gauge Calibration, Non-Programmable Transicoil EMIS Gauge, above. If pointers are out of prescribed range, vibrate the tanks by bumping their lower surfaces. Repeat Fuel Quantity Gauge Calibration, Non-Programmable Transicoil EMIS Gauge, above, if the pointers do not return to the prescribed range.

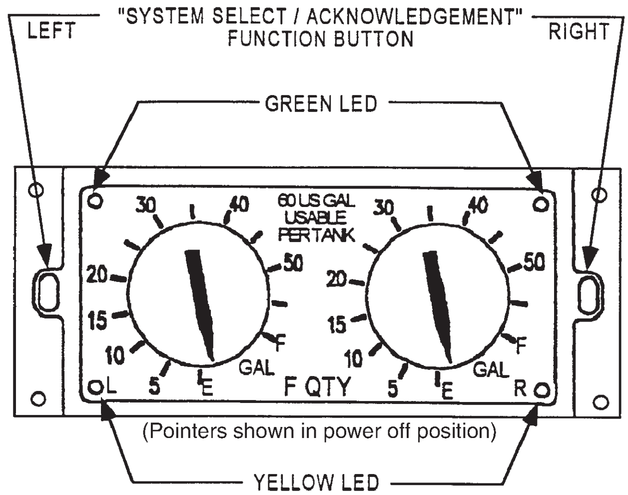
(3) When pointers are adjusted correctly, fill each tank with 5 gals of fuel (after (1) gal. unusable has been added) and record data in “Chart 4”.

(4) Fill tanks in 15, 30, 45 and 60 gal. increments and record data in “Chart 4”. Vibrate the tanks by bumping their lower surfaces at each calibration point.

(a) While filling the tanks, check operation of imbalance fuel annunciator light by adding 10 +1/-0 gals more to right tank than left. Light should illuminate.

(b) Reverse procedure and fill left tank with 10 +1/-0 gals more fuel than right. Again, light must illuminate. (Tanks do not need to be drained for this check). Sixty seconds required from time imbalance exists until light illuminates.

(5) Finish filling tanks to 60 gals capacity each. Tolerance on gauge readings at each increment between “E” and “F” but not including “E” and “F” is ± 2 gallons.



Fuel Quantity Module Dual Analog Indicator
Figure 8

(6) After refueling verify that needles on analog gauges are pointing to the areas of the “F” graduations as described in Fuel Quantity Gauge Calibration, Non-Programmable Transicoil EMIS Gauge, above.

C. Programmable Transicoil EMIS Gauge (S/N’s 4636132 thru 4636374).

(1) Defuel both fuel tanks. As fuel level decreases to 5 +0/-1 gals, verify the low fuel alarms sound.

(2) When defueling is complete, verify that the needles on analog gauges are pointing to the areas of the “E” graduations as shown in “Figure 4” when tanks are empty. Tolerance on “E” (empty indication) is ± 0 gallons. If pointers are out of prescribed range, vibrate the tanks by bumping their lower surfaces. If pointers are out of the prescribed range after vibrating tanks, one or both senders in the tank in question may have to be replaced.

(3) When pointers are indicating correctly, fill each tank with 1 gal (unusable) fuel. Indicators must read “E” as described in “Figure 4”. Record data in “Chart 4”.

(4) Fill tanks in 5, 15, 30, 45 and 60 gal. increments and record data in “Chart 4”. Vibrate the tanks by bumping their lower surfaces at each calibration point.

(a) While filling the tanks, check operation of imbalance fuel annunciator light by adding 10 +1/-0 gals more to right tank than left. Light should illuminate.

(b) Reverse procedure and fill left tank with 10 +1/-0 gals more fuel than right. Again, light must illuminate.

(5) Finish filling tanks to 60 gals capacity each. Tolerance on gauge readings at each increment between “E” and “F”, but not including “E” and “F” is ± 2 gallons. “E” and 1 gal unusable have a tolerance of ± 0 gal. “F” has tolerance of +0/-1 gal.

(6) After refueling verify that needles on analog gauges are pointing to the areas of the “F” graduations as described in “Figure 5”.

D. Complete procedure as follows:

(1) If used, disconnect the digital ohmmeter.

(2) Disconnect the external power supply unit.

(3) Replace access covers.

CHART 4
FUEL CALIBRATION WORKSHEET

FUEL CALIBRATION					LEFT TANKS RIGHT TANKS
	RESISTANCE (OHMS) 1 CONNECTOR P336 **				
TANK CAPACITY (GALLONS)	LEFT PINS ***	RIGHT PINS ***	GAUGE READINGS 2 ANALOG	DIGITAL	TOLERANCE (GALS) (ON GAUGE READING)
0.0	7.1	6.4			±0
(1) Gal. Unusable	7.1	6.4			±0
5	17.9	19.3			±2
15	42.6	41.7			±2
30	64.2	60.9			±2
45	81.5	80.2			±2
60	109.4	108.1			+0, -1

** This connector is located behind instrument.

*** For pin locations, refer to Fuel Quantity Electrical Schematic (S/N's 4636021 and up) in Chapter 91-28-40.

1 - Reference only - not a required measurement. Data applicable to S/N's 4636021 thru 4636131 only.

2 - Allowed 2 gallon spread between analog and digital readings except at "0", "(1) gal unusable" and "Full". Digital and analog must read the same at "0", "(1) gal unusable" and "Full".

6. Fuel Quantity Indicator Calibration (Avidyne Entegra)

(PIR-PPS60226, Rev. A.) (PIR-PPS60231, Rev. B.)

A. Setup

(1) Level the airplane per 8-20-00.

(2) Drain the fuel tanks per 12-10-00, Fuel System, Draining Fuel System - Complete.

B. From the MFD Maintenance page, depress the Line Select Key (L4) to access the Engine Setup page.

C. Depress the “Fuel Cal” button (L1) to enter Fuel Calibration page.

(1) Depress the “Begin Cal” button (R3) and follow the on-screen calibration procedure for left and right tank zero fuel (1.0 + .1-0 gal. unusable) and full calibration (60 + 0-1 gal. usable) points. Wait approximately 3 minutes for fuel level to stabilize.

(2) Use the Right Knob to select the current calibration point. The selected calibration point is highlighted and the value displayed is the current reported fuel quantity from the DAU. A message at the bottom of the screen prompts the operator to add the appropriate amount of fuel and then to select the “Accept Value” button (R3) once the value reported from the DAU has stabilized (wait approximately 3 minutes).

NOTE: If the DAU reported value is not within 2.5 gallons of the test point value, a message “DAU reported Fuel Quantity Out of Tolerance” will be presented and the value will not be accepted.

(3) Once all points have been calibrated, the operator presses “Calibration Complete” to cause the calibration factors to be computed and applied to the DAU reported fuel quantity.

(4) Other options from the Calibration Underway state are to “Restore Last Cal” and “Clear Cal”.

(a) Pressing “Restore Last Cal” causes the calibration values from the last completed calibration to be restored and the state to change to Calibrated.

(b) Pressing “Clear Cal” causes all calibration values to be cleared and the state to change to “Not Calibrated”. An “Are You Sure?” prompt will give the operator a chance to reconsider the decision to either “Restore Last Cal” or “Clear Cal”.

(5) When the calibration procedure has been completed, press the Save button. If you decide not to save the changes, pressing the Cancel button from the Underway state causes the current calibration session to be aborted with any unsaved interim calibration values being discarded. Changed will not take effect until the MFD has been restarted.

NOTE: The “empty” and “full” fuel readings on this setup page will NOT display zero (0) or 60 gallons usable fuel for respective empty and full calibration points. However, the appropriate zero and full (60 U.S. gallons) will be displayed on the MFD in normal operation mode (not maintenance setup.)

(6) Verify DAU Status Box reads: DAU(s) configured for Piper PA-46-350P (Mirage).

(7) Verify Calibration Status Box reads:

“Left Fuel Tank: Calibrated”

“Right Fuel Tank: Calibrated”

7. Fuel Quantity Indicator (Garmin G1000/G1000 NXi)

In PA-46-350P S/N's 4636460, 4636463 and up less 4636481; and PA-46R-350T S/N's 4692134–4692214, less 4692141 4692149 and 4692153.

The fuel quantity indication is incorporated into the G1000/G1000 NXi Multi-Function Display (MFD). The raw data from the 4 fuel senders (2 per side) is provided to the GEA which outputs fuel quantity and diagnostic information to the GIAs. The data then is fed through the respective PFDs and displayed on the MFD.

If a sender or wire failure were to occur, the GEA sends an invalid data bit to the G1000/G1000 NXi system which in-turn will apply a RED-X (G1000) or YELLOW-X (G1000 NXi) across the fuel indicator on the affected tank side.

8. Fuel Quantity Indicator Calibration (Garmin G1000)

(PIR-PPS55026, Rev. J; 107977 F.)

Standard equipment in PA-46-350P S/N's 4636460, 4636463–4636715, 4636717–4636719; and PA-46R-350T S/N's 4692134–4692214, less 4692141 4692149 and 4692153.

The Fuel Quantity Indication is incorporated into the G1000 Multi-Function Display (MFD). The raw fuel quantity data from the fuel senders is sent to the #1 GIA. The data then is fed through the #1 PFD and displayed on the MFD.

A. Setup

(1) Level the airplane per 8-20-00.

(2) Drain the fuel tanks per 12-10-00, Fuel System, Draining Fuel System - Complete.

B. Fuel Tank Calibration

NOTE: If the squat switch is deactivated due to jacking of aircraft, pull the hydraulic pump and landing gear indicator circuit breakers prior to performing the fuel tank calibration.

NOTE: The large FMS Knob changes Groups and the small FMS knob changes Pages within a Group.

(1) Turn on power to the aircraft.

(2) Start the #1 PFD & #2 PFD in the configuration mode by holding the ENT key while applying power. Start the MFD while holding the far right softkey while applying power.

NOTE: The G1000 System must be powered for three minutes before proceeding.

(3) Add unusable fuel to both wings (1 US gal. per side).

(4) After 5 minutes, select the 'CAL' group, then select 'FUEL CALIBRATION' page on 'PFD #1'.

(5) Unlock the page by pressing softkeys 12 (far right softkey), 11, 10, 9 in sequence.

(6) Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

(7) Press the 'RAW DATA' softkey to de-select the scale function.

(8) Press the 'EMPTY' softkey. Press the 'ENT' key to acknowledge the prompt after the 'EMPTY' softkey is pressed.

NOTE: If the 'EMPTY' softkey is not active, 0.5 US gal. (1.5 US gal. unusable fuel) may be added to activate the 'EMPTY' softkey and document with QA.

(9) Verify the 'ACTUAL QUANTITY' shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 0 gallons.

(10) Press the 'TNK SEL' softkey.

(11) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(12) Press the 'ENT' key.

(13) Repeat items (6) thru (9) for the right tank.

NOTE: For actual fuel quantity entries, the large FMS knob moves the cursor. The small FMS knob changes the digits.

(14) Add 15 gal fuel to both wings.
(15) After at least 5 minutes, Verify the ‘CALIBRATION VALUE’ indication for the Right Tank is stable before proceeding.
(16) Press the ‘FUEL QTY’ softkey.
(17) Enter 15 gal for ‘ACTUAL FUEL QUANTITY’. Press the ‘ENT’ key to accept.
(18) Press ‘ENT’ Key again to calibrate.
(19) Verify the ‘ACTUAL QUANTITY’ shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 15 gallons.
(20) Press the ‘TNK SEL’ softkey.
(21) Rotate the small FMS knob to generate a pick list and select the Left Tank.
(22) Press the ‘ENT’ key.
(23) Repeat items (15) thru (19) for the Left Tank.
(24) Add 15 gal fuel to both wings.
(25) After at least 5 minutes, Verify the ‘CALIBRATION VALUE’ indication for the Left Tank is stable before proceeding.
(26) Press the ‘FUEL QTY’ softkey.
(27) Enter 30 gal for ‘ACTUAL FUEL QUANTITY’. Press the ‘ENT’ key to accept.
(28) Press ‘ENT’ Key again to calibrate.
(29) Verify the ‘ACTUAL QUANTITY’ shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 30 gallons.
(30) Press the ‘TNK SEL’ softkey.
(31) Rotate the small FMS knob to generate a pick list and select the Right Tank.
(32) Press the ‘ENT’ key.
(33) Repeat items (25) thru (29) for the Right Tank.
(34) Add 15 gal fuel to both wings.
(35) After at least 5 minutes, Verify the ‘CALIBRATION VALUE’ indication for the Right Tank is stable before proceeding.
(36) Press the ‘FUEL QTY’ softkey.
(37) Enter 45 gal for ‘ACTUAL FUEL QUANTITY’. Press the ‘ENT’ key to accept.
(38) Press ‘ENT’ Key again to calibrate.
(39) Verify the ‘ACTUAL QUANTITY’ shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 45 gallons.
(40) Press the ‘TNK SEL’ softkey.
(41) Rotate the small FMS knob to generate a pick list and select the Left Tank.
(42) Press the ‘ENT’ key.
(43) Repeat items (35) thru (39) for the Left Tank.
(44) Add 15 gal more fuel to both wings.

NOTE: If the full 15 gallons cannot be added, note the quantity of fuel that was added to the tank until full to the filler neck.

(45) After at least 5 minutes, Verify the ‘CALIBRATION VALUE’ indication for the Left Tank is stable before proceeding.

(46) Press the ‘FULL’ softkey. Press the ‘ENT’ key to acknowledge the prompt after the ‘FULL’ softkey is pressed.

(47) Verify the ‘ACTUAL QUANTITY’ shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 60 gallons.

(48) Press the ‘TNK SEL’ softkey.

(49) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(50) Press the ‘ENT’ key.

(51) Repeat items (45) thru (47) for the Right Tank.

C. Full Range Accuracy Check

NOTE: If an aircraft fails the fuel accuracy check, troubleshoot to determine cause and corrective action and repeat the fuel calibration and accuracy check, if required.

(1) Level the airplane per 8-20-00.

(2) Drain the fuel tanks per 12-10-00, Fuel System, Draining Fuel System - Complete.

(3) Turn on power to the aircraft.

(4) Add unusuable fuel to both wings (1 US gal per side) and allow to stabilize (5 min.).

(5) Verify the Garmin fuel indicator shows left and right fuel quantities at 0 gal.

(6) Verify fuel quantity gauge digits are RED and “FUEL QTY LOW” warning CAS message is displayed.

(7) Add 10 gal of fuel to both wings and verify the corresponding reading on the Garmin system is within +/- 4 gal per side (+/- 8 gal total).

(8) Verify that the fuel quantity gauge digits turn from RED to YELLOW and “FUEL QTY LOW” warning CAS message is extinguished when fuel quantity is greater than 5 gal of fuel per side.

(9) Continue adding fuel in 10 gal increments (total of 5 increments) and verify the corresponding reading on the Garmin system is within +/- 4 gal per side (+/- 8 gal total).

(10) Verify fuel quantity gauge digits turn WHITE upon reaching greater than 10 gallons of fuel per side.

(11) If quantity readings at all intervals are within limits, then fuel accuracy check is complete. Turn aircraft power OFF.

9. Fuel Quantity Indicator Calibration (Garmin G1000 NXi)

(PIR-107998 F.)

Standard equipment in PA-46-350P S/N’s 4636716, 4636720 and up.

The Fuel Quantity Indication is incorporated into the G1000 NXi Multi-Function Display (MFD). The raw fuel quantity data from the fuel senders is sent to the #1 GIA. The data then is fed through the #1 PFD and displayed on the MFD.

A. Setup

(1) Level the airplane per 8-20-00.

(2) Drain the fuel tanks per 12-10-00, Fuel System, Draining Fuel System - Complete.

B. Fuel Tank Calibration

NOTE: If the squat switch is deactivated due to jacking of aircraft, pull the hydraulic pump and landing gear indicator circuit breakers prior to performing the fuel tank calibration.

NOTE: The large FMS Knob changes Groups and the small FMS knob changes Pages within a Group.

(1) Turn on power to the aircraft.

(2) Start the #1 PFD, MFD, & #2 PFD in the configuration mode by holding the ENT key on the PFDs, and the far right softkey on the MFD while applying power.

NOTE: The G1000 NXi System must be powered for three minutes before proceeding.

(3) Select the CAL group, then select FUEL TANK CALIBRATION page on the PFD #1.

(4) Unlock the page by pressing softkeys: 12 (far right softkey), 11, 10, 9 in sequence.

(5) Verify sensors inputs values are less than the following (If value is higher, then there is an issue with the probe or the wiring):

(a) LEFT 1: 8.2

(b) LEFT 2: 12.3

(c) LEFT 3: 7.0

(6) Press Tank SEL softkey, rotate the small FMS knob to generate a pick list and select RIGHT Tank. Press the ENT key.

(7) Repeat item “(5)” for the right tank (RIGHT instead of LEFT for sensors inputs).

(8) Add unusable fuel to both wings (1 US gal. per side).

(9) Rotate the small FMS knob to generate a pick list and select the LEFT Tank. Press the ENT key.

(10) After at least 5 minutes, verify the CALIBRATION VALUE indication for the Left Tank is stable before proceeding.

(11) Press the RAW DATA softkey to de-select the scale function.

(12) Press the EMPTY softkey. Press the ENT key to acknowledge the prompt after the EMPTY softkey is pressed.

NOTE: If the EMPTY softkey is not active, 0.5 US gal. (1.5 US gal. unusable fuel) may be added to activate the EMPTY softkey.

(13) Verify the “ACTUAL QUANTITY” shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 0 gallons.

(14) Press the Tank SEL softkey, rotate the small FMS knob to generate a pick list and select the RIGHT Tank. Press the ENT key.

(15) Repeat items “(10)” thru “(13)” for the Right Tank.

NOTE: For actual fuel quantity entries, the large FMS knob moves the cursor. The small FMS knob changes the digits.

(16) Add 15 gal fuel to each wing.

(17) After at least 5 minutes, verify the CALIBRATION VALUE indication for the Right Tank is stable before proceeding.

(18) Press the FUEL QTY softkey.

(19) Enter 15 gal for ACTUAL FUEL QUANTITY. Press the ENT key to accept.

(20) Press ENT Key again to calibrate.

(21) Verify the “ACTUAL QUANTITY” shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 15 gallons.

(22) Press the Tank SEL softkey, rotate the small FMS knob to generate a pick list and select the LEFT Tank. Press the ENT key.

(23) Repeat items “(17)” thru “(21)” for the Left Tank.

(24) Add 15 gal fuel to each wing.

(25) After at least 5 minutes, verify the CALIBRATION VALUE indication for the Left Tank is stable before proceeding.

(26) Press the FUEL QTY softkey.

(27) Enter 30 gal for ACTUAL FUEL QUANTITY. Press the ENT key to accept.

(28) Press ENT Key again to calibrate.

(29) Verify the “ACTUAL QUANTITY” shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 30 gallons.

(30) Press the Tank SEL softkey, rotate the small FMS knob to generate a pick list and select the RIGHT Tank. Press the ENT key.

(31) Repeat items “(25)” thru “(29)” for the Right Tank.

(32) Add 15 gal fuel to each wing.

(33) After at least 5 minutes, verify the CALIBRATION VALUE indication for the Right Tank is stable before proceeding.

(34) Press the FUEL QTY softkey.

(35) Enter 45 gal for ACTUAL FUEL QUANTITY. Press the ENT key to accept.

(36) Press ENT Key again to calibrate.

(37) Verify the “ACTUAL QUANTITY” shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 45 gallons.

(38) Press the Tank SEL softkey, rotate the small FMS knob to generate a pick list and select the LEFT Tank. Press the ENT key.

(39) Repeat items 34 thru 38 for the Left Tank.

(40) Add 15 gal fuel to each wing.

NOTE: If 15 gallons cannot be added, note the fuel quantity added to the tank until full to the filler neck.

(41) After at least 5 minutes, verify the CALIBRATION VALUE indication for the Left Tank is stable before proceeding.

(42) Press the FULL softkey. Press the ENT key to acknowledge the prompt after the FULL softkey is pressed.

(43) Verify the “ACTUAL QUANTITY” shown in the “CALIBRATION TABLE” in the lower right corner of the screen shows 60 gallons.

(44) Press the Tank SEL softkey, rotate the small FMS knob to generate a pick list and select the RIGHT Tank. Press the ENT key.

(45) Repeat items “(41)” thru “(43)” for the Right Tank.

C. Full Range Accuracy Check

If an aircraft fails the fuel accuracy check, troubleshoot to determine cause and corrective action and repeat the fuel calibration and accuracy check, if required.

(1) Level aircraft and ensure both wing tanks are completely empty.

(2) Turn on power to the aircraft.

(3) Add unusable fuel to both wings (1 US gal per side) and allow to stabilize (5 min.).

(4) Verify the Garmin fuel indicator shows left and right fuel quantities at 0 gal.

(5) Add 10 gal of fuel to both wings and verify the corresponding reading on the Garmin system is within +/- 4 gal per side (+/- 8 gal total).

(6) Continue adding fuel in 10 gal increments (total of 6 increments) and verify the corresponding reading on the Garmin system is within +/- 4 gal per side (+/- 8 gal total).

(7) If quantity readings at all intervals are within limits, then fuel accuracy check is complete. Turn aircraft power OFF.

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CHAPTER

29

HYDRAULIC POWER

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CHAPTER 29

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
29-List of Effective Pages	1	Mar 25/21			
	2	Sep 15/09			
29-Table of Contents	1	Mar 25/21			
	2	Sep 15/09			
29-00-00	1	Nov 30/17			
	2	Nov 30/17			
29-10-00	1	Dec 15/14			
	2	Sep 15/09			
	3	Dec 15/14			
	4	Dec 15/14			
	5	Dec 15/14			
	6	Sep 15/09			
	7	Sep 15/09			
	8	Mar 25/21			
	9	Dec 15/14			
	10	Mar 25/21			
	11	Mar 25/21			
	12	Mar 25/21			
	13	Nov 30/17			
	14	Mar 25/21			
	15	Mar 25/21			
	16	Mar 25/21			
	17	Mar 25/21			
	18	Mar 25/21			

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CHAPTER 29 - HYDRAULIC POWER

TABLE OF CONTENTS

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GENERAL

1. Description and Operation

The hydraulic system components covered in this chapter consist of the combination hydraulic pump/ reservoir and hydraulic lines. The brake system, although hydraulically operated, is not included in this section as it has its own hydraulic system independent of the gear system. The brake system and landing gear are covered in Chapter 32. The flap system in this aircraft is electric and is covered in Chapter 27.

WARNING: PRIOR TO CHECKING OUT THE HYDRAULIC SYSTEM, PLACE THE AIRPLANE ON JACKS. (REFER TO JACKING, 7-10-00.)

The hydraulic system provides the power to operate the landing gear using an electrically driven hydraulic pump operating at 1350 to 1400 maximum psi. The hydraulic pump is located behind the aft baggage compartment with access through a removable panel. The pump has a fluid sight gauge for ease of checking fluid level, as an integral part of the unit. A relief valve in the pump prevents system pressure from exceeding 1600 psi during pump operation. A thermal relief valve prevents hydraulic pressure from exceeding 2600 psi when the landing gear is down and locked and airplane is on the ground.

The landing gear selector is an electrical switch that commands the pump to operate in either the up or down position. The gear is held in the up position by hydraulic pressure. Emergency gear extension is accomplished by a free fall function of the gear selector valve which releases gear up hydraulic pressure, allowing the gear to extend and lock.

2. Troubleshooting

If trouble develops in landing gear, place the airplane on jacks (refer to Jacking, 7-10-00), and proceed to find the cause of the trouble. A hydraulic system operational check is provided under Hydraulic System Testing, 29-10-00. See also Landing Gear Retraction System Functional Test, 32-30-00. When the trouble has been recognized, the first step is to isolate the cause. After isolation of a particular cause or problem, refer to the individual Chapter pertaining to that subject matter, i.e. Landing Gear: Chapter 32.

Hydraulic system troubles are not always traceable to one cause. It is possible that a malfunction may be the result of more than one trouble in the system. Start with the most obvious and most probable reasons for the trouble, check each possibility and, by process of elimination, isolate the troubles.

In PA-46-350P S/N's 4636299, 4636314 and up; and, all PA-46R-350T airplanes: Landing gear extension and retraction issues can frequently be traced to faulty up/down pressure switches in the hydraulic pump. See “Troubleshooting” on page 29108.

3. Hydraulic System Leading Particulars

See “Chart 1” and “Chart 2” on page 29002.

CHART 1
HYDRAULIC SYSTEM LEADING PARTICULARS (PARKER HANNIFIN)

Hydraulic Pump, Electrically Driven, Reversible	
Flow Rate @ 1350 psi	1 gpm
Hydraulic Fluid	MIL-PRF-5606
Normal Operating Pressure	1350 psi
Relief Valve (Thermal)	Opens at 2600 +0 /- 50 psig
Relief Valve (Pressure)	Opens at 1600 +0 /- 50 psig
Pressure Switch	
Open	1350 - 1400 psig max.
Close	1170 psig min.
Electrical Characteristics	
Voltage	24 - 28 Vdc
Operating Current @ 1350 psig	55 Amps Max. @ 27 Vdc
Operating Time	Gear Up 7 - 8 sec
	Gear Down 7 - 8 sec
Overload Protection	25 Amp Circuit Breaker

CHART 2
HYDRAULIC SYSTEM LEADING PARTICULARS (FRISBY/TRIUMPH)

Hydraulic Pump, Electrically Driven, Reversible	
Flow Rate @ 1350 psi	1 gpm
Hydraulic Fluid	MIL-PRF-5606
Reservoir Capacity	30 cu. in. (.49 liters)
Normal Operating Pressure	1350 psi
Relief Valve (Thermal)	Opens at 2525 ± 75 psig
Relief Valve (Pressure)	Opens at 1600 +0 /- 50 psig
Pressure Switch	
Open	1350 - 1400 psig max.
Close	1170 psig min.
Electrical Characteristics	
Voltage	24 - 28 Vdc
Operating Current @ 1350 psig	55 Amps Max. @ 27 Vdc
Operating Time	Gear Up 7 - 8 sec
	Gear Down 7 - 8 sec
Overload Protection	25 Amp Circuit Breaker

MAIN

1. Description

These airplanes are equipped with an electrically driven, reversible hydraulic pump. The pump is mounted on a floor bracket just forward of the aft pressure bulkhead.

2. Hydraulic Pump (Parker Hannifin)

S/N's 4636001 thru 4636313 only. (See "Figure 3".)

WARNING: PLACE AIRPLANE ON JACKS PRIOR TO WORKING ON THE HYDRAULIC SYSTEM. (REFER TO 7-10-00.)

A. Removal

The hydraulic pump is located on a floor bracket just forward of the aft pressure bulkhead assembly.

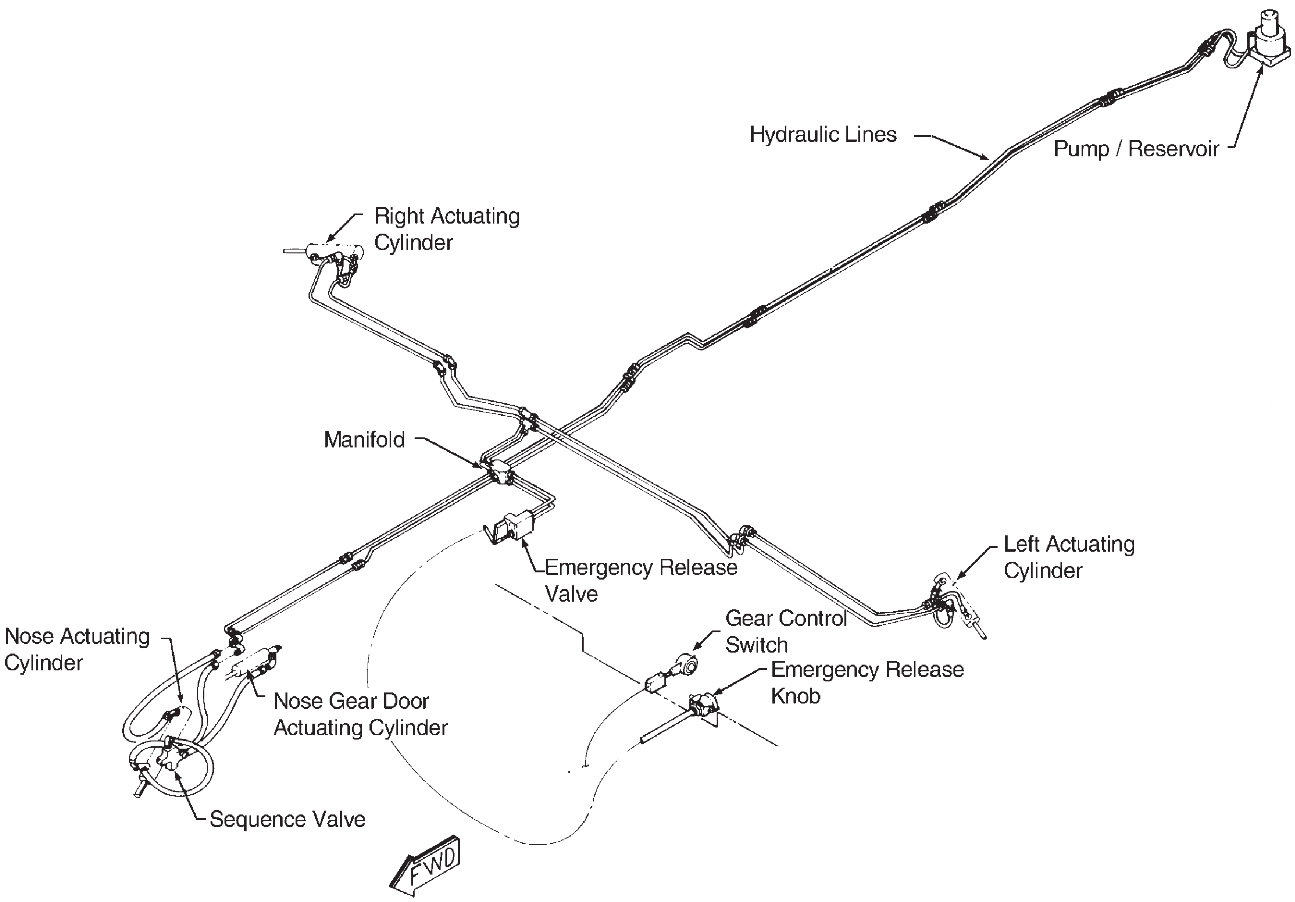
(1) Place airplane on jacks (refer to 7-10-00, Jacking).

(2) Reduce hydraulic pressure as follows:

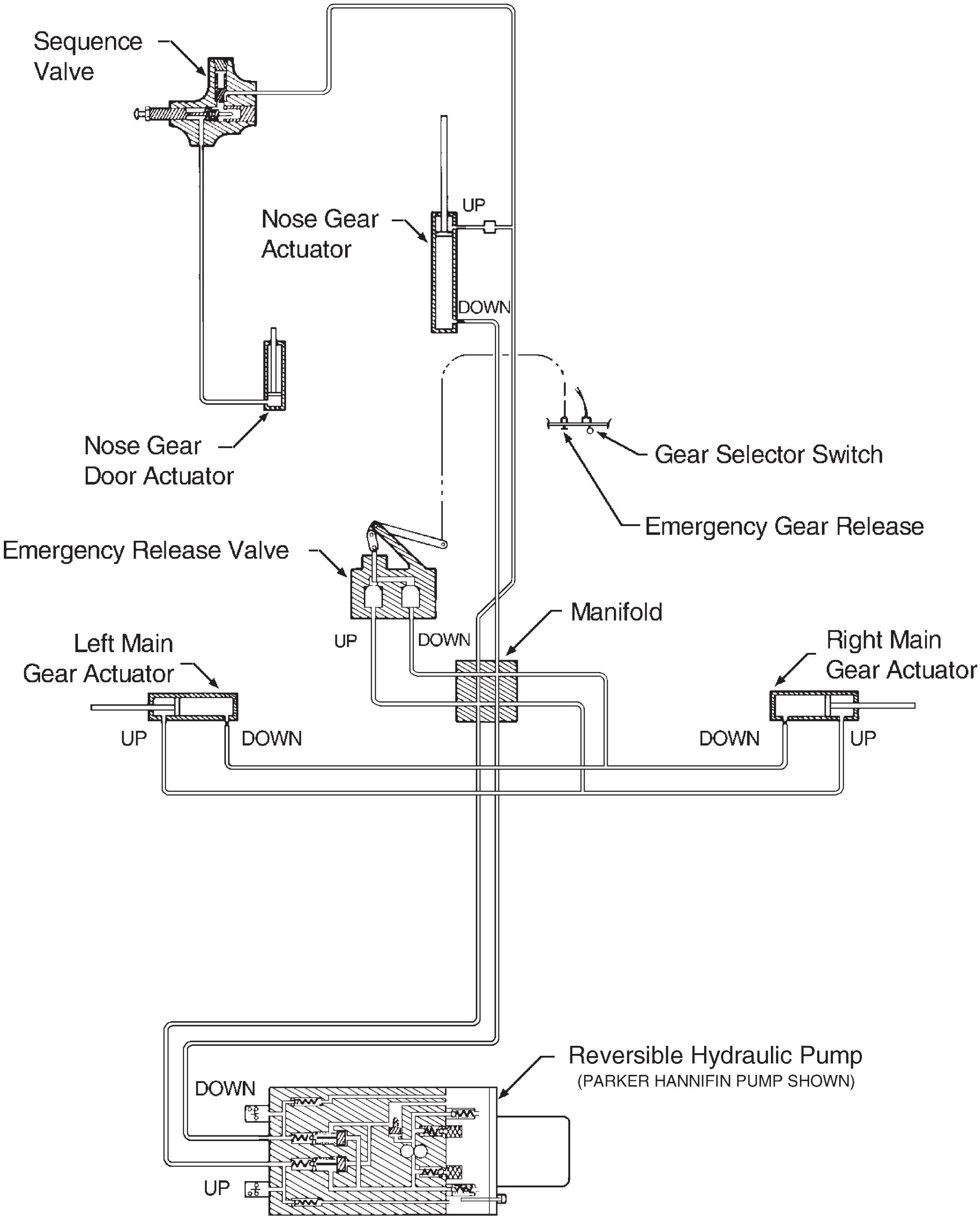
(a) Place gear switch in up position.

(b) Allow gear to retract until pressure drops.

(c) Pull hydraulic pump circuit breaker.



Hydraulic System Installation
Figure 1



Fluid Diagram of Hydraulic System
Figure 2

(3) Set BATT MASTER switch to OFF position.

(4) Disconnect aft baggage net assembly.

(5) Remove access panel aft of baggage compartment.

(6) Disconnect electrical wires and tie to pump.

(7) Disconnect and rapidly cap hydraulic lines and hoses.

(8) Remove four sets of mounting bolts, washers, rubber washers, and nuts.

(9) Remove pump from airplane to a clean bench.

(10) Drain reservoir into a suitable container.

B. Installation

WARNING: PLACE AIRPLANE ON JACKS PRIOR TO WORKING ON THE HYDRAULIC SYSTEM. (REFER TO 7-10-00.)

(1) Place pump assembly in aircraft.

NOTE: Align pump assembly so that sight glass is visible.

(2) Install four sets of mounting bolts, washers, and rubber washers.

(3) Uncap and reconnect hydraulic lines.

(4) Fill reservoir with hydraulic fluid.

(5) Set BATT MASTER switch on.

(6) Reset hydraulic pump circuit breaker.

(7) Cycle landing gear in accordance with Landing Gear Retraction Functional Test, 32-30-00. Verify that the gear operates smoothly and there are no leaks.

C. Disassembly

See “Figure 3”, “Figure 4”, and “Figure 5”.

NOTE: Pump may not require complete disassembly. Follow steps (1) thru (8) as needed for access to air vent, motor, pump, reservoir, pressure-relief valves, and O-rings. Sight glass may be removed (step (4)) without removing other parts.

NOTE: Follow steps (9) thru (13) as needed (prior steps not necessary) for removal of pressure switches, elbow/restrictor fittings, upper manifold, and lower manifold.

NOTE: Drain pump prior to disassembly. Mark pump assembly along length for ease in aligning parts.

(1) Remove air vent by removing lockwire, bolt, and washer. Discard O-ring.

NOTE: Hold cap plate down while removing hardware to prevent spillage of any remaining hydraulic fluid.

(2) Remove remaining three bolts and washers from cap plate.

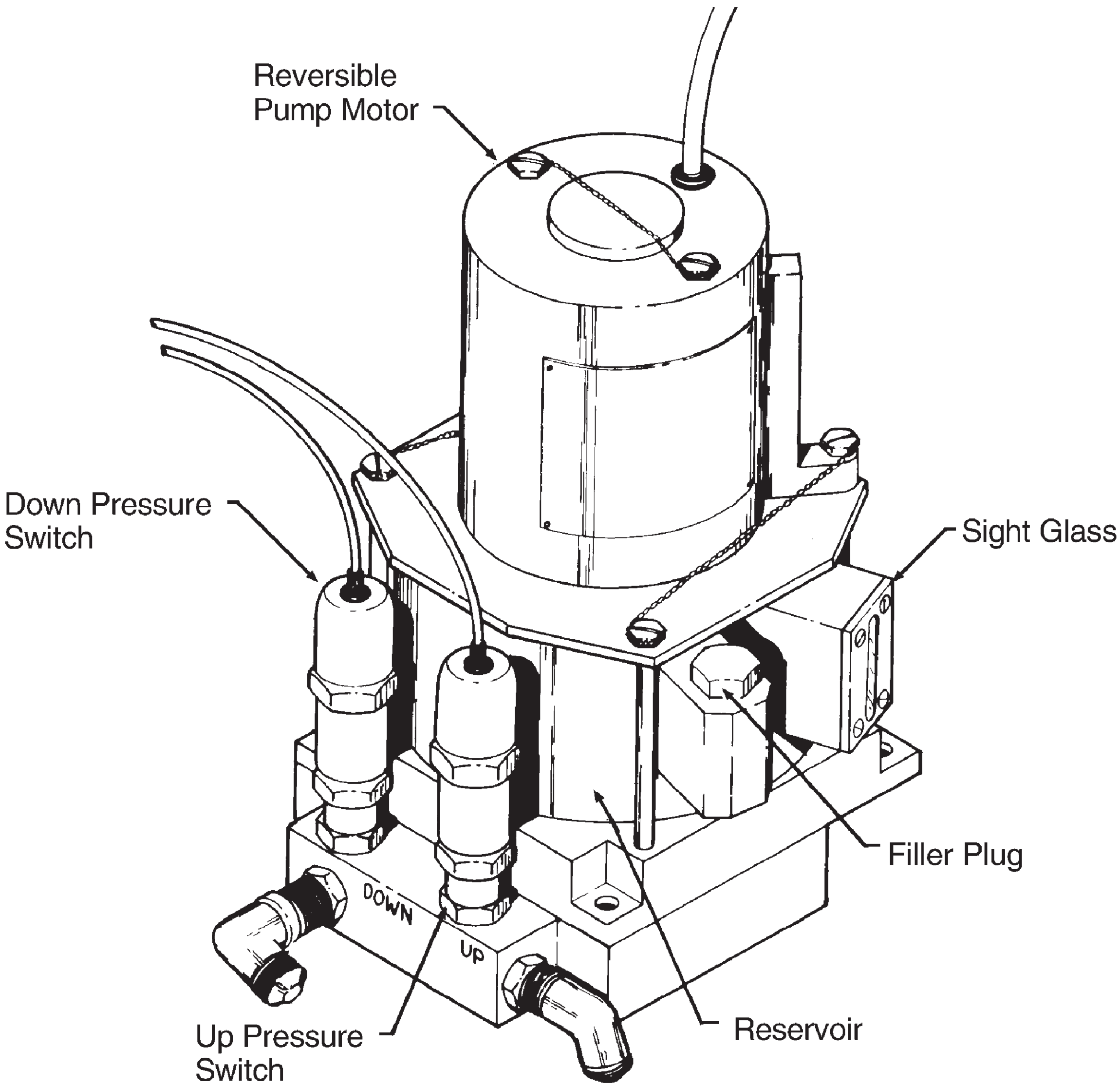
(3) Remove coupler, O-ring, motor and cap plate. Motor and cap plate remain as one unit.

NOTE: If necessary, separate motor from cap plate by removing screws. Check and replace shaft seal on cap plate as needed. Do not disassemble motor.

(4) Remove reservoir. If necessary, remove sight glass by removing four screws and gasket. Discard gasket.

NOTE: Perform steps (5), (6), (7), or (8) only as needed for parts beneath reservoir: pump, filters, O-rings, and pressure-relief valves.

(5) Remove pump by removing lockwire, four screws, and four washers that secure oil deflector and pump.



Effectivity
4636001 thru 4636313

Hydraulic Pump (Parker Hannifin)
Figure 3

(6) Remove intake filters by removing retaining rings.

(7) Remove small O-rings under filters and large O-ring on upper manifold.

(8) Remove pressure relief valves from upper manifold.

NOTE: Follow steps (9) thru (13) as needed to remove pressure switches and manifolds.

(9) Remove pressure switches. Discard O-rings.

(10) Remove elbow and restrictor fitting by loosening nut(s), unscrewing elbow(s) from lower manifold, and discarding O-ring(s).

NOTE: Upper and lower manifolds must be replaced as a set in pump serial numbers 156 and up. These can be distinguished by gold color instead of gray in earlier models. Refer to PA-46-350P Parts Catalog.

(11) Remove and discard large O-ring from upper manifold if not already performed in step (10).

(12) Separate upper and lower manifolds, if necessary, by removing lockwire, four screws, and washers.

(13) Remove and discard O-rings between upper and lower manifolds.

D. Assembly

See “Figure 3”, “Figure 4”, and “Figure 5”.

WARNING: USE MEK IN WELL-VENTILATED AREA. KEEP AWAY FROM FLAME OR HEAT. AVOID CONTACT WITH SKIN OR EYES.

CAUTION: DO NOT SOAK ELECTRICAL PARTS.

CAUTION: DO NOT REUSE LOCKWIRE OR O-RINGS. COAT REPLACEMENT O-RINGS WITH HYDRAULIC FLUID.

(1) Check and replace parts as needed. If necessary, clean with MEK or other suitable solvent.

(2) Replace O-rings between upper and lower manifolds. Secure manifolds with screws and washers. Torque 45–55 in.-lbs. in a crossing pattern in four steps: 20, 30, 40 and 50 in.-lbs. Install lockwire.

(3) Install pump, oil deflector, four screws, and washers. Take care in aligning. (See “Figure 4”, and “Figure 5”.) Torque screws 10–14 in.-lbs. Secure with lockwire.

(4) Install pressure relief valves and O-rings. Torque 65–70 in.-lbs.

(5) Replace O-rings at intake filters. Install intake filters. Secure with retaining rings.

(6) If replacing sight glass, install gasket, glass, and cover onto reservoir with four screws. Torque 8–12 in.-lbs.

(7) Install O-ring onto upper manifold.

(8) Install reservoir. Align carefully on manifold. (See “Figure 4”, and “Figure 5”)

(9) If cap plate and motor have been separated:

(a) Install new shaft seal onto cap plate.

(b) Assemble cap plate to motor with four screws, washers, and seals.

(c) Torque 25–35 in.-lbs.

CAUTION: DO NOT ALLOW LOCKWIRE TO INTERFERE WITH MOTOR SHAFT.

(d) Lockwire screws. (See “Figure 5”.)

(10) Install O-ring on cap plate assembly.

(11) Install coupler onto hydraulic pump shaft.

(12) Align cap plate bolts with threaded inserts of upper manifold. Lift plate and turn coupler so that slot aligns with motor shaft.

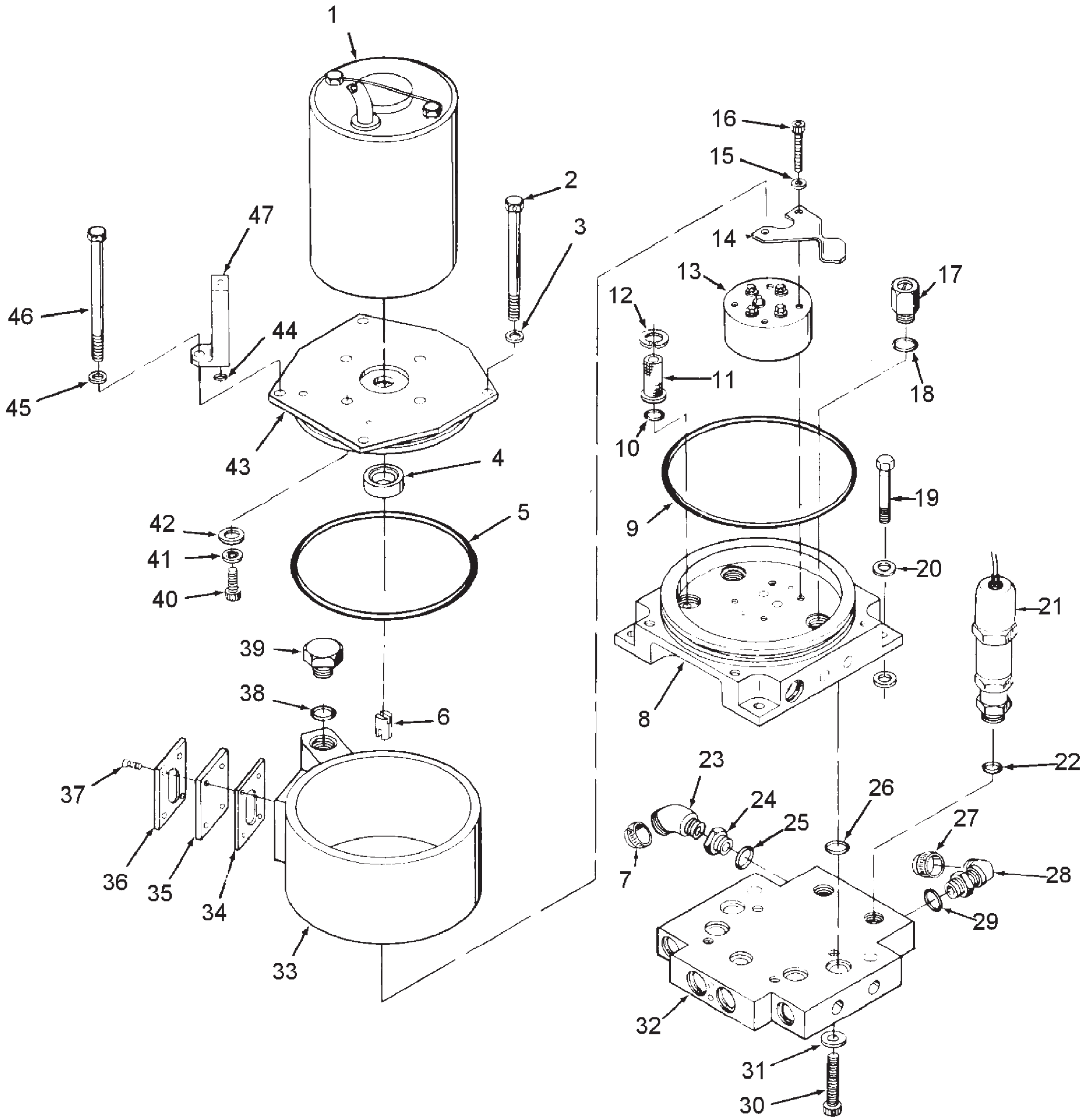
(13) Install cap plate assembly onto reservoir so that coupler and motor shaft engage. Align bolt holes with threaded inserts.

(14) Secure with bolts and washers. Torque 25–35 in.-lbs. Secure with lockwire.

(15) Install O-ring in air vent assembly. Secure air vent to cap plate assembly with bolt and washer. Torque 25–35 in.-lbs. Secure with lockwire.

(16) Replace O-rings for pressure switches. Install pressure switches. Torque pressure switches 65–70 in.-lbs.

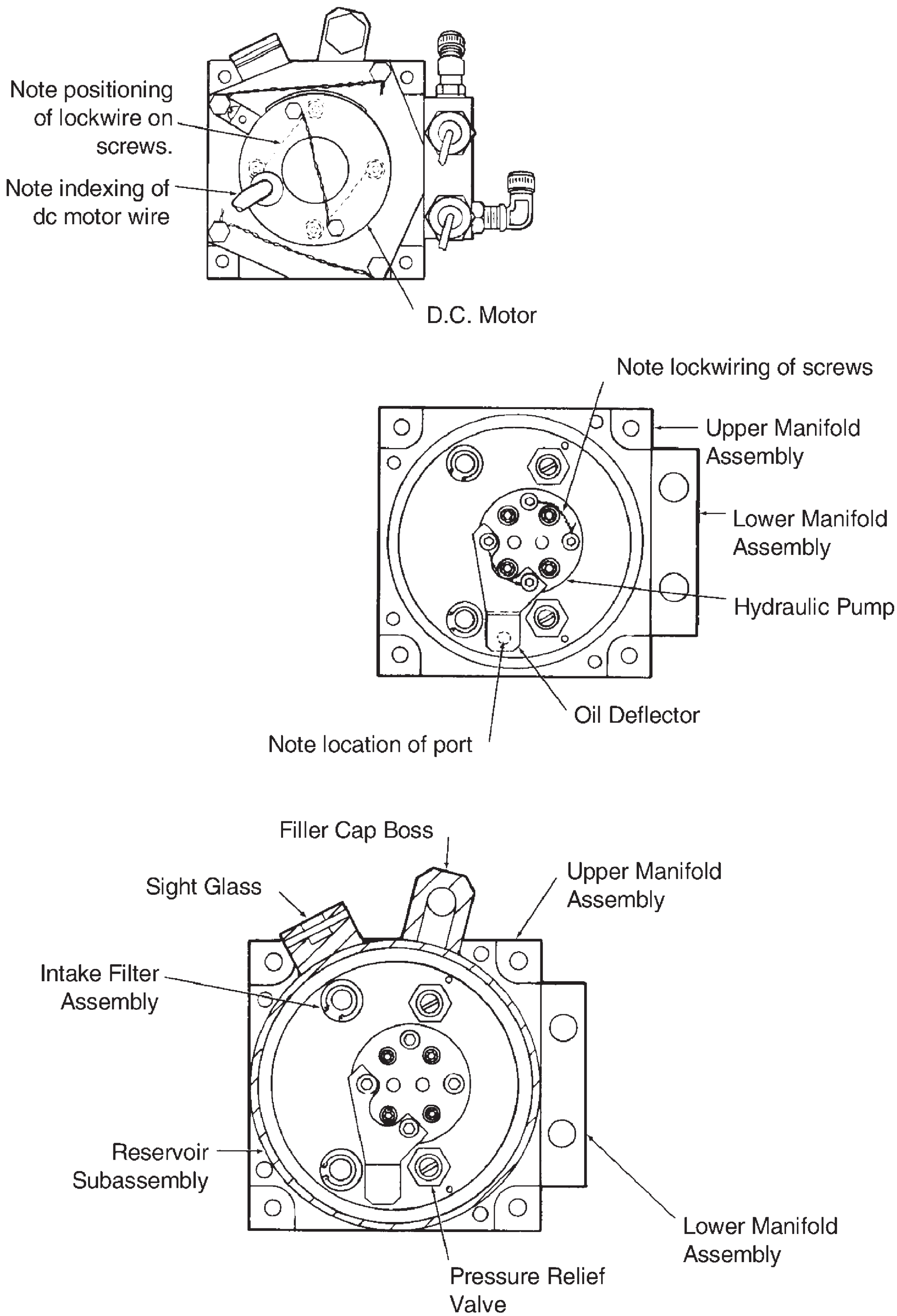
(17) Install O-rings for elbow fittings. Thread elbow fittings into UP and DOWN ports in lower manifold so that fittings point downward. Torque nut to 65–70 in.-lbs.



1. D.C. MOTOR
2. BOLT
3. WASHER
4. SHAFT SEAL
5. O-RING
6. COUPLER
7. PROTECTIVE CAP
8. UPPER MANIFOLD
9. O-RING
10. O-RING
11. INTAKE FILTER
12. RETAINING RING
13. HYDRAULIC PUMP
14. OIL DEFLECTOR
15. WASHER
16. SCREW
17. PRESSURE RELIEF VALVE
18. O-RING
19. MOUNTING BOLT
20. WASHER
21. PRESSURE SWITCH
22. O-RING
23. 45° ELBOW
24. NUT
25. O-RING
26. O-RING
27. PROTECTIVE CAP
28. 90° ELBOW
29. O-RING
30. SCREW
31. WASHER
32. LOWER MANIFOLD
33. RESERVOIR
34. GASKET
35. SIGHT GLASS
36. SIGHT GLASS COVER
37. SCREW
38. O-RING
39. FILLER CAP
40. SCREW
41. WASHER
42. STAT-O-SEAL
43. CAP PLATE
44. O-RING
45. WASHER
46. BOLT
47. AIR VENT ASSEMBLY

Effectivity
4636001 thru 4636313

Hydraulic Pump (Parker Hannifin) - Disassembled
Figure 4



Hydraulic Pump (Parker Hannifin) - Subassembly Alignment
Figure 5

Effectivity
4636001 thru 4636313

3. Hydraulic Pump (Frisby/Triumph)

S/N's 4636314 and up; S/N's 4692001 and up. See "Figure 6" on page 29109.

CAUTION: THE HYDRAULIC PUMP IS NOT SERVICEABLE IN THE FIELD. REPLACE WITH NEW OR FACTORY REBUILT UNIT. RETURN DEFECTIVE AND OVERLIMIT UNITS TO THE FACTORY FOR REBUILD.

A. Troubleshooting

CAUTION: NOT COMPLETING THE APPLICABLE TROUBLESHOOTING PROCEDURE COULD LEAD TO MISDIAGNOSIS OF THE PROBLEM. REVIEW AND COMPLETE TROUBLESHOOTING PRIOR TO CONTINUING.

NOTE: This section incorporates requirements from Piper Service Letter No 1222B.

These instructions apply to the following hydraulic pumps (Piper part number (Piper code number)): 102559-003 (461-768); 102559-004 (461-770).

Troubleshoot the landing gear extension and retraction system per Section 32-30-00 of the applicable Piper airplane maintenance manual (AMM). If troubleshooting indicates that the hydraulic pump is the likely cause of the trouble, determine if any of the following symptoms are present:

- The pump runs continuously
- The pump starts and stops continuously
- The landing gear fails to retract or extend

If any of these symptoms are present, remove the pressure switches and bench test them per "Pressure Switch Test", before replacing the pump.

B. Pressure Switch Test

(1) Required Equipment

(a) Hydraulic Pressure Calibration Set (commercially available): Must have a working pressure range of 0–2,000 psig minimum. See "Figure 7" on page 291010.

(b) Ohmmeter.

(c) Hydraulic Fluid MIL-PRF-5606 at a sufficient quantity to operate hydraulic pressure calibration set.

(2) Set-up

WARNING: INHALATION OF HYDRAULIC FLUID OIL MIST OR VAPORS AT ELEVATED TEMPERATURES MAY CAUSE RESPIRATORY IRRITATION. THEY CAN BE HARMFUL IF ASPIRATED INTO THE LUNGS.

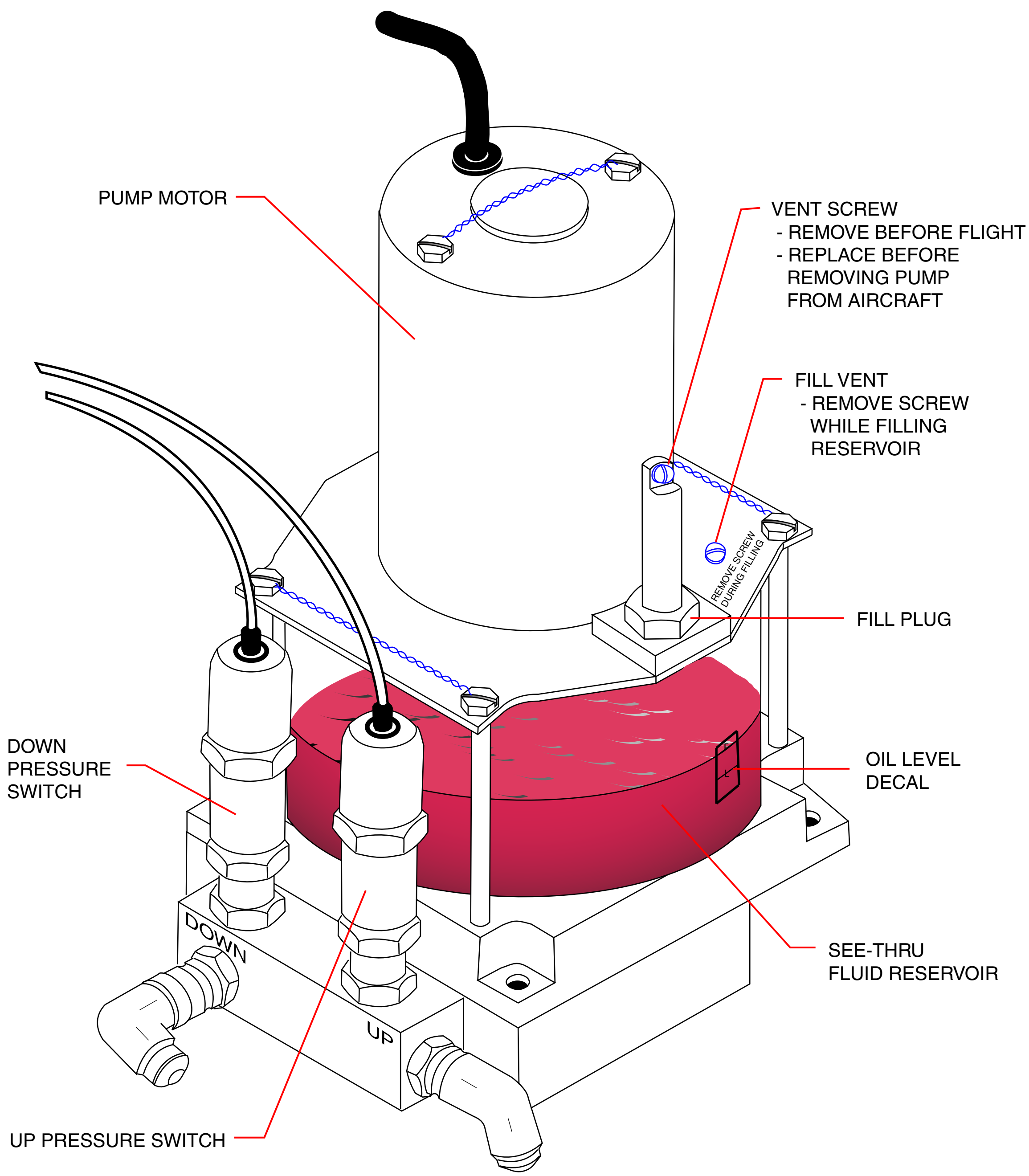
(a) Test fluid is to be in accordance with MIL-PRF-5606.

(b) Ambient Temperature at test must be 60 to 100°F (16 to 38°C).

(c) Ambient Humidity at test must be 10 to 90%.

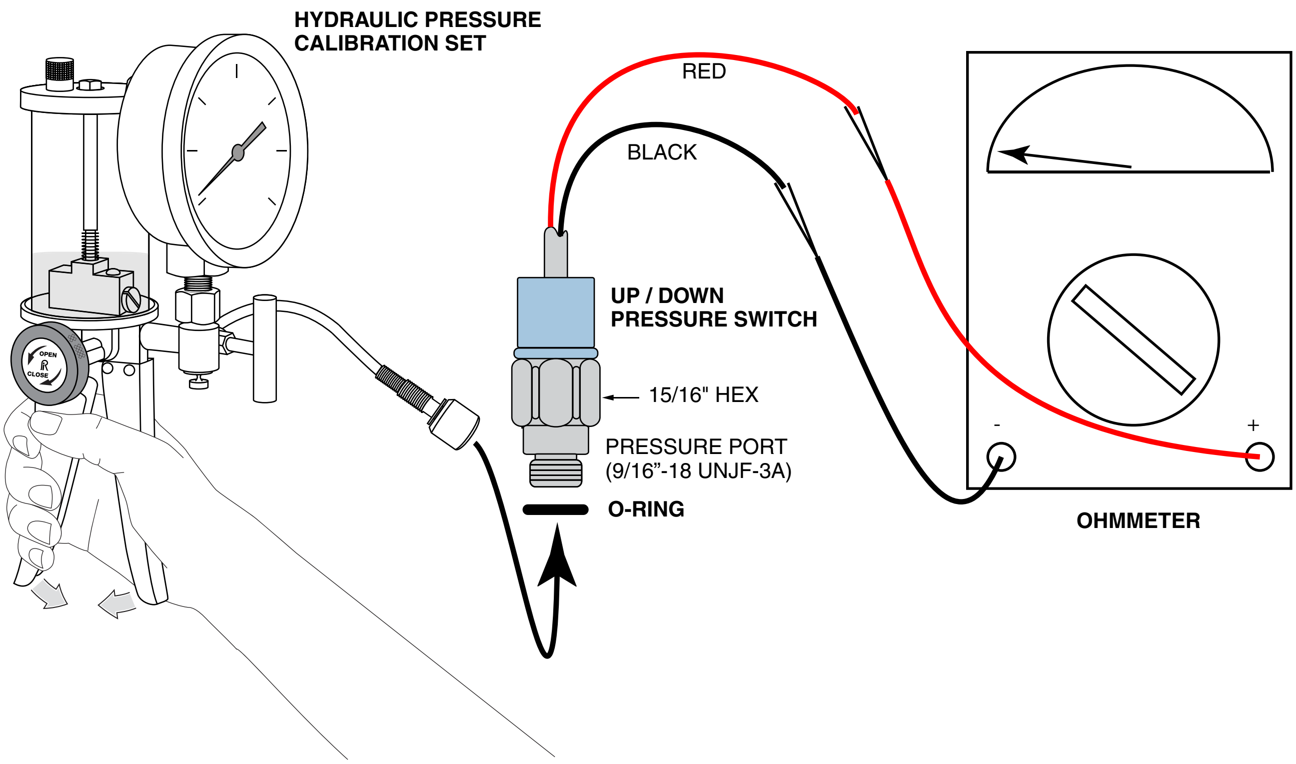
(d) Fluid Temperature at test must be 70 to 110°F (21 to 43°C).

102559 B



Hydraulic Pump (Frisby/Triumph)
Figure 6

Effectivity
4636314 and up
4692001 and up



Pressure Switch Test Setup
Figure 7

(3) Test Procedure

(a) Remove switches from the pump per "Removal" on page 291012.

(b) Lubricate a new O-ring with hydraulic fluid and install it on the pressure switch.

(c) Connect the pressure switch to a hydraulic pressure calibration set and an ohmmeter as shown in "Figure 7" on page 291010.

NOTE: The hydraulic pressure calibration set shown in "Figure 7" on page 291010 is a Ralston HPGV series unit, but any commercially available set meeting the minimum operating range per "Required Equipment", above, is acceptable.

(d) Bleed the hydraulic pressure calibration set per the manufacturer's instructions.

(e) Set the ohmmeter ON and observe that the switch is closed (i.e., ohmmeter shows continuity).

(f) Test each pressure switch as follows:

NOTE: Operate the calibration set per the manufacturer's instructions.

1) Increase the pressure to the pressure switch slowly until the switch opens (i.e., ohmmeter loses continuity).

The pressure switch must open at:

a) 1450 ± 40 psig (100.0 ± 2.8 bar) for Triumph part number (P/N) 1FA1305939-1.

b) 1350 ± 40 psig (93.1 ± 2.8 bar) for Triumph P/N 1FA1305939.

2) Slowly reduce pressure to the switch until it closes (i.e., ohmmeter shows continuity). The pressure switch must close before 1170 psig (80.7 bar).

3) Repeat Steps 1) and 2) five times to confirm the consistent operation of the switch.

4) Test results:

NOTE: If both pressure switches pass the test, then they are not the problem. The hydraulic pump may be faulty, but troubleshooting other system components independently (such as the free fall valve) is recommended.

a) If a pressure switch passes the test, reinstall it in the pump per "Installation" on page 291012.

b) If a pressure switch fails to open and close as specified above, it must be replaced. Proceed to "Pressure Switch Replacement" on page 291012.

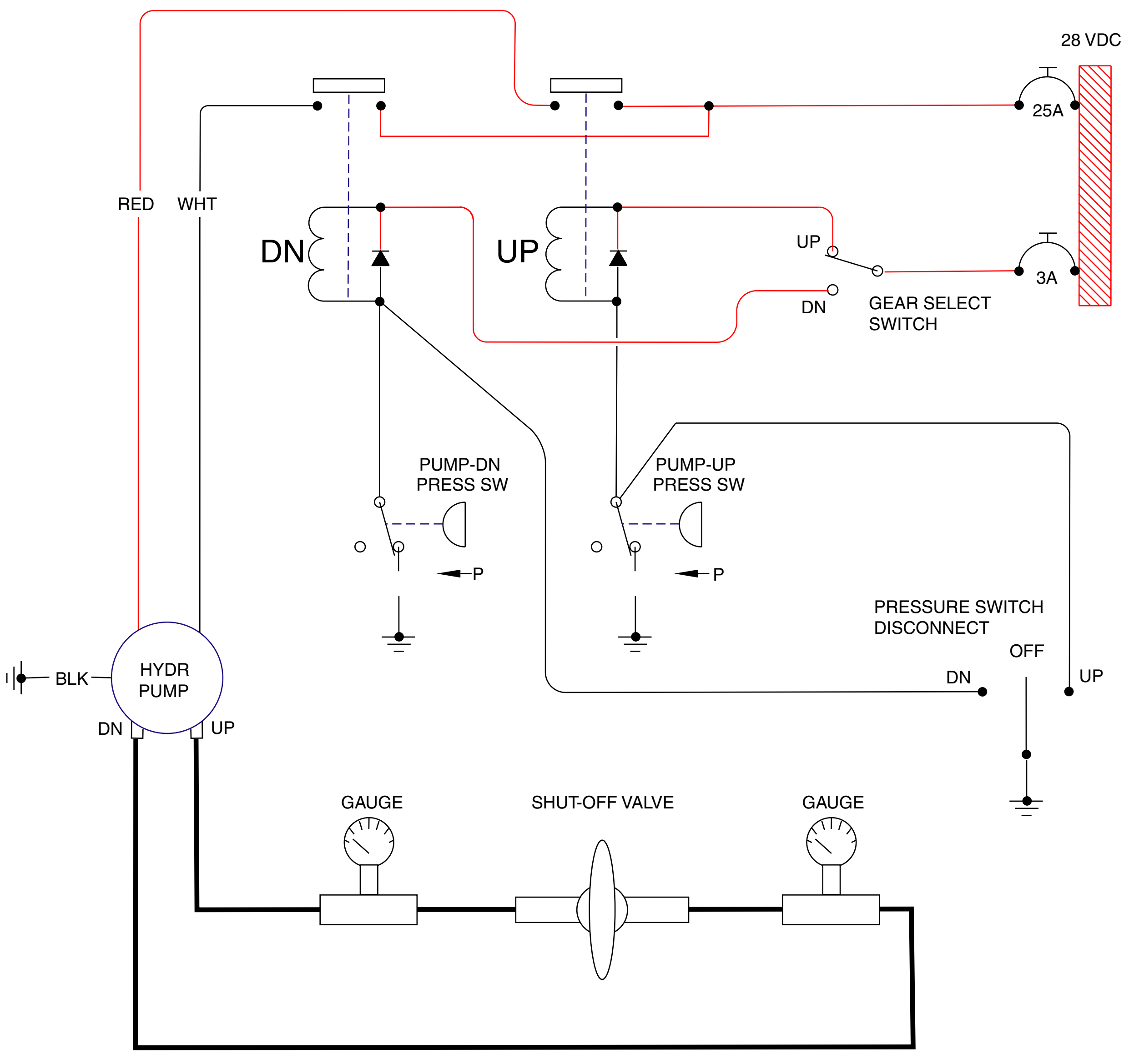
(4) If it appears that the hydraulic pump is faulty, prior to replacing it, repeat Step (2) "Troubleshooting", to confirm if the problem is in the hydraulic pump or elsewhere in the system.

C. Pressure Switch Replacement

NOTE: The replacement hydraulic pump pressure switch (Piper P/N 767-238) available from Piper is Triumph P/N 1FA1305939-1. Both switches must be the same Triumph part number. Early airplanes may be equipped with hydraulic pumps that have Triumph P/N 1FA1305939 pressure switches. If so, both switches must be replaced by Piper P/N 767-238 (Triumph P/N 1FA1305939-1) switches, even if only one is faulty.

The following procedures assume the hydraulic pump is in the airplane. Not all of the steps are required if the pump is already on a test bench.

(1) Removal
  (a) Place the airplane on jacks per Section 7-10-00.
  (b) Reduce hydraulic pressure as follows:
    1) Place the gear switch in the UP position.
    2) Allow the gear to retract until pressure drops.
    3) Pull OUT the hydraulic pump circuit breaker.
    4) Set the master switch to the OFF position.
  (c) Drain the hydraulic fluid from the pump by removing the fill plug on the reservoir and siphoning pump reservoir dry.
  (d) Disconnect the pressure switch electrical leads.
  (e) Unscrew the pressure switch from pump.
  (f) Discard the O-ring.
(2) If required, test the switch per “Pressure Switch Test” on page 29108.
(3) Installation
  (a) Lubricate a new O-ring with hydraulic fluid and install it on the new (or found good) pressure switch.
  (b) Install the pressure switch in the pump. Torque to 110–115 in-lb (11.3–13.0 N-m).
  (c) Verify that the airplane is on jacks.
  (d) Verify that the master switch is set to the OFF position.
  (e) Connect the electrical leads to the pressure switch.
  (f) Connect a 28 Vdc regulated power source capable of supplying a minimum of 50 amperes to the aircraft electrical system.
  (g) Set the master switch to the ON position. Push IN the hydraulic pump circuit breaker.
  (h) Refill the pump with hydraulic fluid, then bleed the system per “Bleeding Hydraulic System (Frisby/Triumph)” on page 291017.
  (i) Conduct the appropriate Landing Gear Retraction System Functional Test in 32-30-00 to verify proper operation.



Hydraulic Pump (Frisby/Triumph) Test Set-up
Figure 8

Effectivity
4636314 and up
4692001 and up

D. Removal

WARNING: PLACE AIRPLANE ON JACKS PRIOR TO WORKING ON THE HYDRAULIC SYSTEM. (REFER TO 7-10-00)

The hydraulic pump is located on a floor bracket just forward of the aft pressure bulkhead assembly.

(1) Place airplane on jacks (refer to 7-10-00, Jacking).
(2) Reduce hydraulic pressure as follows:
  (a) Place gear switch in up position.
  (b) Allow gear to retract until pressure drops.
  (c) Pull hydraulic pump circuit breaker.
(3) Set BATT MASTER switch to OFF position.
(4) Disconnect aft baggage net assembly.
(5) Remove access panel aft of baggage compartment.
(6) Disconnect electrical wires and tie to pump.
(7) Disconnect and rapidly cap hydraulic lines and hoses.
(8) Install vent screw, washer, and warning tag.
(9) Remove four sets of mounting bolts, washers, rubber washers and nuts.
(10) Remove pump from airplane to a clean bench.
(11) Drain reservoir into a suitable container.

E. Installation

WARNING: BE SURE AIRPLANE IS ON JACKS BEFORE BLEEDING AND TESTING HYDRAULIC SYSTEM.

(1) Place pump assembly in aircraft.

  NOTE: Align pump assembly so that fill plug is to the front.

(2) Install four sets of mounting bolts, washers, and rubber washers.
(3) Uncap and reconnect hydraulic lines.
(4) Fill reservoir with hydraulic fluid.

  NOTE: Do not overfill. Add fluid only when level falls below the “L” mark. When filling, allow fluid level to stabilize before comparing fluid level to fill marks. Overfilling will result in excess fluid being expelled from reservoir vent pipe.

(5) Remove vent screw, washer and warning tag.
(6) Set BATT MASTER switch on.
(7) Reset hydraulic pump circuit breaker.
(8) Bleed hydraulic system per Bleeding Hydraulic System, below.
(9) Conduct Landing Gear Retraction System Functional Test, 32-30-00. Verify that the gear operates smoothly and there are no leaks.

F. Testing

(1) Normal Operation Test

See “Figure 8” on page 291013.

Connect the hydraulic pump into a system as shown in “Figure 8” on page 291013. to check normal pump operation. Maintain 27-28 Vdc during all testing.

(a) Fill pump reservoir to full line (“F” mark on the reservoir) with clean filtered MIL-PRF-5606 hydraulic fluid. Ensure that pump vent is open (remove screw, washer and warning tag, if installed). With the shut-off valve open, run the pump in the gear “UP” position for approximately 10 seconds. Slowly close the valve to build up 50-100 psig, and bleed air from the “UP” side of the system. Refill reservoir as required to keep fluid level between “F” and “L” marks during all testing.

NOTE: During all tests, add clean filtered MIL-PRF-5606 (Revision “F” or higher) hydraulic fluid to the reservoir as necessary to keep fluid level just above the “L” mark on the sight gauge with the landing gear in the down and locked position. Verify that the fluid level does not exceed the “F” mark on the reservoir when the landing gear is in the up and locked position. Do not overfill. Add fluid only when level falls below the “L” mark. When filling, allow fluid level to stabilize before comparing fluid level to fill marks. Overfilling will result in excess fluid being expelled from reservoir vent pipe.

(b) With the shut-off valve in the open position, apply power to the pump with the gear switch in the gear “UP” position. Allow the pump to operate approximately 15 seconds. Slowly close the shut-off valve. The pressure switch shall turn the pump off at 1400 psig maximum. Continue to supply electrical power to the pump and observe for pressure drop and no pump operation for 5 minutes. Except, if the pump operates once momentarily during this five minute period there shall be no repeated pump operation during the subsequent fifteen minutes.

(c) Repeat (b), above, with the gear switch in the gear “DOWN” position.

(d) With the shut-off valve open, the gear selector switch in the gear “DOWN” position and the pressure switch disconnect switch in the “DOWN” position, allow the pump to operate approximately 10 seconds. Slowly close the shut-off valve. Observe the maximum pressure indicated and immediately shut pump off. The maximum pressure shall be at least 1550 psig but not more than 1800 psig. With the pump off observe that the pressure does not fall below 1350 psig in 5 minutes.

(e) Repeat (4), above, with the gear switch and pressure switch disconnect switch in the gear “UP” position. The maximum pressure shall be at least 1550 psig but not more than 2000 psig.

NOTE: If pump is not installed in an aircraft, reinstall vent screw, washer, and warning tag upon completion of test.

(2) Thermal Relief Valve Operation Test

(a) Fill pump reservoir to full line (“F” mark on the reservoir) with clean filtered MIL-PRF-5606 hydraulic fluid. Ensure that pump vent is open (remove screw, washer and warning tag, if installed).

(b) With a suitable pump and calibrated gauge, slowly apply hydraulic pressure individually to the pump “UP” and “DOWN” outlet ports. The first indication of pressure relief shall occur at 2450 to 2600 psig for both “UP” and “DOWN” sides of the pump assembly.

NOTE: If pump is not installed in an aircraft, reinstall vent screw, washer, and warning tag upon completion of test.

4. Servicing Hydraulic System

(PIR-PPS60033-7, Rev. G/-9, Rev. P/-13, Rev. F.)

WARNING: BE SURE AIRPLANE IS ON JACKS BEFORE BLEEDING AND TESTING HYDRAULIC SYSTEM.

A. Setup

(1) Place the airplane on jacks per 7-10-00.

(2) Connect a 28 Vdc regulated power source capable of supplying a minimum of 50 amperes to the aircraft electrical system.

NOTE: With power supplied through the auxiliary power plug, the battery-master switch should remain in the “OFF” position. The APU circuit bypasses the battery-master switch to energize the ship power circuits.

NOTE: Before starting test, check for adequate clearance of nose gear lights, sequence valves, actuators, hoses, etc.

NOTE: Never operate the emergency gear release piston with the hydraulic pump circuit breaker “IN”.

(3) During all tests, add clean filtered MIL-PRF-5606 (Revision “F” or higher) hydraulic fluid to the reservoir as necessary to keep fluid level just above the “L” mark on the sight gauge with the landing gear in the down and locked position. Verify that the fluid level does not exceed the “F” mark on the reservoir when the landing gear is in the up and locked position.

B. Bleeding Hydraulic System (Parker Hannifin)

S/N’s 4636001 thru 4636313 only.

CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE HYDRAULIC PUMP IS OPERATING. ALWAYS PULL THE HYDRAULIC PUMP CIRCUIT BREAKER BEFORE TESTING THE EMERGENCY GEAR RELEASE SYSTEM. FAILURE TO COMPLY WILL DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

CAUTION: DO NOT OVERFILL AND DO NOT ALLOW HYDRAULIC FLUID LEVEL TO DROP BELOW THE “L” (LOW) MARK ON THE SIGHT GAUGE WITH THE GEAR IN THE DOWN POSITION.

NOTE: Follow this sequence below to reduce foaming. Add fluid to reservoir as needed.

(1) Disconnect rod assemblies between nose gear doors and door actuating torque tube assembly to prevent door actuation.

(2) Screw the sequence valve plunger adjustment screw all the way IN.

(3) With the gear down, pull OUT the emergency gear release knob.

(4) Place the gear selector in the UP position. Operate pump for 10 to 15 seconds.

(5) Return the gear selector to the DOWN position. Push the emergency gear release knob IN.

(6) Operate the gear through at least five (5) complete cycles, adding fluid to the reservoir as required.

(7) Starting with the gear down, pull the hydraulic pump circuit breaker and place the gear selector switch in the UP position.

(8) Using the circuit breaker as a switch, allow the gear to retract until the nose gear is about half way up and then pull the circuit breaker.

(9) Loosen the “B” nut on hose assembly at nose gear door actuator. Depress plunger on sequence valve to purge air from line.

NOTE: Move nose gear downward if necessary to create sufficient pressure and flow to completely bleed air from the line.

Repeat as necessary until all air is removed and then retighten the “B” nut.

(10) Loosen the “B” nuts at the main gear UP fittings. Move the main gears downward to bleed air from the actuators. Tighten the “B” nuts.

(11) Push IN the circuit breaker and complete the retraction cycle.

(12) Place gear selector in DOWN position and loosen “B” nut on DOWN fittings on all three gear actuators to bleed any trapped air. Retighten “B” nuts.

NOTE: Pump may run intermittently during bleeding procedure.

(13) With the gear down, pull OUT the emergency gear release knob.

(14) Place the gear selector in the UP position. Operate pump for 10 to 15 seconds.

(15) Return the gear selector to the DOWN position. Push the emergency gear release knob IN.

(16) Operate the landing gear through at least three (3) complete cycles.

C. Bleeding Hydraulic System (Frisby/Triumph)

S/N’s 4636314 and up; S/N’s 4692001 and up.

CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE HYDRAULIC PUMP IS OPERATING. ALWAYS PULL THE HYDRAULIC PUMP CIRCUIT BREAKER BEFORE TESTING THE EMERGENCY GEAR RELEASE SYSTEM. FAILURE TO COMPLY WILL DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

CAUTION: IN S/N’S 4636314 AND UP AND S/N’S 4692001 AND UP, DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE LANDING GEAR ARE DOWN AND LOCKED. DOING SO CAN DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

CAUTION: DO NOT OVERFILL AND DO NOT ALLOW HYDRAULIC FLUID LEVEL TO DROP BELOW THE “L” (LOW) MARK ON THE SEE-THRU RESERVOIR OIL LEVEL DECAL WITH THE GEAR IN THE DOWN POSITION.

(1) Without power applied and with landing gear down, fill reservoir to top.

(2) Ensure gear selector is in the DOWN position. Connect power to aircraft. With 25A circuit breaker initially pulled out, use it as a momentary switch while watching fluid level in reservoir. Do not allow fluid to drop below “L” (low) mark. Refill reservoir to top and repeat until the gear motor shuts off with 25A circuit breaker pushed “in” which indicates the down side of system is full.

(3) Add fluid to between “L” (low) and “F” (full) marks on the see-thru reservoir.

(4) Ensure 25A circuit breaker is out. Place gear selector in the UP position. Use 25A circuit breaker as a momentary switch while watching fluid level in reservoir. Refill reservoir as necessary and continue until gear motor shuts off with the 25A circuit breaker pushed “in”.

(5) Leave gear up while working on gear doors etc. Allow 10–30 minutes for air to clear from system. Monitor fluid level in reservoir and add as necessary.

(6) Lower landing gear. Check door fit etc. and then run the gear up and down 3–4 times.

(7) Check fluid level in reservoir and add as necessary. Air should now be out of the system.

(8) Operate the landing gear through at least three (3) complete cycles.

5. Hydraulic Lines - Removal and Installation

   A. Remove damaged hydraulic lines by disconnecting the fittings at each end and where it is secured by brackets. See “Figure 1” on page 29101 as an aid in locating the attaching brackets and bends in the lines.

   B. Provide a small container for draining the line.

   C. Install new or repaired line in the reverse order of removal.

   D. Bleed system and lines.

   E. Check fluid level in reservoir.

6. Hydraulic System Testing

WARNING: BE SURE AIRPLANE IS ON JACKS BEFORE BLEEDING AND TESTING HYDRAULIC SYSTEM.

WARNING: SET BATT MASTER SWITCH TO THE OFF POSITION BEFORE INSERTING OR REMOVING EXTERNAL POWER SUPPLY PLUG.

The hydraulic system should be tested to check that it functions properly after performing any service or repairs. It is suggested that the airplane be connected to an outside power source in order to conserve the battery. See also Landing Gear Retraction System Functional Test, 32-30-00.

   A. Place airplane on jacks (refer to 7-10-00).

   B. With the gear down, place the BATT MASTER switch to the on position, and circuit breaker closed. Place the landing gear selector in the UP position. Pump should immediately operate and gear retract. The red gear unsafe light on the instrument panel should illuminate, until the gear is fully retracted. The hydraulic pump should stop after full gear retraction.

   C. Place the gear selector in the DOWN position. The gear should extend and lock. The gear down lights on the instrument panel will illuminate when all three gears are locked in the down position. Inspect hydraulic system for leakage.

   D. Recycle landing gear to check that it functions properly.

   WARNING: PRIOR TO REMOVING THE AIRPLANE FROM THE JACKS, TURN MASTER SWITCH ON AND VERIFY THAT ALL THREE GREEN LIGHTS ARE ILLUMINATED. THIS WILL INDICATE THE LANDING GEAR IS DOWN AND LOCKED.

   E. Remove airplane from jacks.

7. Hydraulic Pump / Reservoir Servicing

Servicing procedures for the hydraulic pump/reservoir, may be found in 12-10-00 of this manual.

CHAPTER

30

ICE AND RAIN PROTECTION

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CHAPTER 30

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
30-List of Effective Pages	1	Mar 25/21
	2	Sep 15/09
30-Table of Contents	1	Mar 25/21
	2	Nov 30/17
30-00-00	1	Sep 15/09
	2	Sep 15/09
30-10-00	1	Sep 15/09
	2	Sep 15/09
	3	Feb 15/14
	4	Sep 15/09
	5	Sep 15/09
	6	Sep 15/09
	7	Jul 1/10
	8	Sep 15/09
	9	Sep 15/09
	10	Mar 25/21
	11	Sep 15/09
	12	Nov 30/17
	13	Nov 30/17
	14	Nov 30/17
	15	Nov 30/17
	16	Nov 30/17
	17	Nov 30/17
	18	Nov 30/17
	19	Nov 30/17
	20	Sep 15/09
	21	Sep 15/09
	22	Nov 30/17
	23	Nov 30/17
	24	Nov 30/17
	25	Nov 30/17
	26	Nov 30/17
	27	Nov 30/17
	28	Nov 30/17
30-30-00	1	Nov 30/17
	2	Sep 15/09
30-40-00	1	Dec 15/14
	2	Sep 15/09
	3	Sep 15/09
	4	Sep 15/09
	5	Sep 15/09
	6	Sep 15/09
	7	Nov 30/17
	8	Nov 30/17
30-60-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Sep 15/09
	5	Nov 30/17
	6	Sep 15/09
	7	Jan 15/13
	8	Jan 15/13
	9	Feb 15/14
	10	Jan 15/13
	11	Nov 30/17
	12	Jan 15/13
	13	Nov 30/17
	14	Nov 30/17
	15	Nov 30/17
	16	Nov 30/17
	17	Nov 30/17
	18	Sep 15/09

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CHAPTER 30 - ICE AND RAIN PROTECTION

TABLE OF CONTENTS

CHAPTER 30 - ICE AND RAIN PROTECTION

TABLE OF CONTENTS (continued)

GENERAL

1. Description and Operation

The ice protection system is made up of six separate systems: an optional pneumatic wing deice system, an optional electrical propeller heat system, an optional electrically-heated windshield, standard heated stall warning, standard heated pitot, and a wing inspection light installation. The systems can be installed individually or in any combination. The systems are controlled from a Deice Switch Panel (S/N's 4636001 thru 4636131) or Environmental / Deice Switch Panel (S/N's 4636132 and up and S/N's 4692001 and up) on the right instrument panel.

The pneumatic system utilizes boots to displace ice from the leading edges of the flight surfaces. The boots are inflated by the vacuum pumps. See 37-10-00 for pump maintenance. Along with the boots and related plumbing, the system also utilizes a flow valve assembly which incorporates three deicer flow valve assemblies, three pressure switches, a check-valve, a vacuum manifold, and a vacuum manifold/check-valve assembly.

Boots are attached to the leading edge of the wings, vertical stabilizer, and elevator. The boots are made up of a fabric-reinforced rubber containing built-in span-wise inflation tubes. The boots are secured to the leading edges with cement and are connected to the plumbing through the skin by flexible or aluminum air connections.

The propeller heat system consists of an electrically heated boot bonded to each blade, a slip ring assembly connected to the hub of the propeller, a timer, a circuit breaker, a control switch located on the Deice Switch Panel (S/N's 4636001 thru 4636131) or Environmental / Deice Panel (S/N's 4636132 and up and S/N's 4692001 and up) ; and, in S/N's 4636001 thru 4636131 only, a dedicated ammeter and shunt. In S/N's 4636132 and up and in S/N's 4692001 and up, the BFG timer is replaced by a Moritz unit.

When installed, the propeller heat system on this airplane uses single-element type heater boots. In this type system each heater has one electrothermal heating element. When the PROP HEAT switch on the instrument panel is turned ON, the timer directs power through the modular brush assembly and slip ring to all the heating elements for approximately 90 seconds. The timer then switches power off for approximately 90 seconds. This cycle will continue as long as the PROP HEAT switch is in the ON position. In S/N's 46360132 and up only, there is a test mode which uses a squat switch relay to turn the propeller heat off if the airplane remains on the ground more than 30-45 seconds after PROP HEAT is switched ON. When the airplane becomes airborne, PROP HEAT exits the test mode and enters in-flight mode.

NOTE: In S/N's 4636001 thru 4636131 only, heating may begin at either phase of the cycle depending on the timer position when the switch was turned off from previous use.

When installed, the heated windshield consists of an electrically heated left windshield. Circuit protection is provided by the windshield heat CONTROL and POWER breakers in the ICE PROTECTION section of the circuit breaker panel.

Heated pitot and stall warning systems are standard on the airplane. These systems are operated by switches on the Deice Switch Panel (S/N's 4636001 thru 4636131) or Environmental / Deice Switch Panel (S/N's 4636132 and up and S/N's 4692001 and up). Other components of the systems include a heated pitot head (one or two) with an accompanying 10-amp circuit breaker and a heated lift detector with a 25 amp circuit breaker.

To aid the pilot in inspecting the leading edge of the left wing for any ice formation, a wing inspection light may be installed in the left hand side of the fuselage. Information on the wing inspection light may be found in Chapter 33 (Lights).

NOTE: For wiring schematics not found in this chapter, refer to Chapter 91.

2. SMARTboot™ Ice Detection System (S/N's 4636132 thru 4636313 only)

The optional pneumatic wing deice system, as installed in S/N's 4636132 thru 4636313 only, incorporates the B.F. Goodrich SMARTboot™ Ice Detection System. This system advises the pilot of icing conditions on the empennage of the airplane. Electrical sensors integrated directly into the surface of the pneumatic de-icers on the vertical fin and right horizontal stabilizer advise the presence of ice and appropriate time to de-ice.

NOTE: The system is installed as and performs only an advisory function.

AIRFOIL

1. Pneumatic Deice System

The pneumatic system utilizes boots to displace ice from the leading edges of the flight surfaces. The boots are inflated by the vacuum pumps. See 37-10-00 for pump maintenance. Along with the boots and related plumbing, the system also utilizes a flow-valve assembly which incorporates three deicer flow valve assemblies, three pressure switches, a check-valve, a vacuum manifold (if installed), and a pressure manifold/check valve assembly.

Boots are attached to the leading edge of the wings, vertical stabilizer, and elevator. The boots are made up of a fabric-reinforced rubber containing built-in, span-wise, inflation tubes. The boots are secured to the leading edges with cement and are connected to the plumbing, through the skin, by flexible or aluminum air connections.

The system is controlled from a control panel on the right side of the instrument panel.

2. Troubleshooting

See “Chart 1”.

3. Components

A. Deice Boots

See Deicers (Boots) (Pneumatic Deice System), below.

B. Deice Flow Valve Assembly

The deice flow valve assembly is located behind the firewall on the right-hand side of the airplane. It is mounted to the lower fuselage.

(1) Removal

(a) Disconnect the electrical leads to the individual deice valves at the (three) connectors.

(b) Disconnect the five hoses from the deice flow valve assembly.

(c) Remove the bolts which secure the deice flow valve assembly to the lower fuselage. Remove the deice flow valve assembly.

(2) Installation

(a) Position the deice flow valve assembly on the lower fuselage and secure it with the bolts.

(b) Reconnect the five hoses to the deice flow valve assembly.

(c) Reconnect the electrical leads from the (three) deice valves.

C. Pressure Control Valve

The pressure control valves are located on the forward right-hand side of the firewall.

(1) Removal

(a) Disconnect the two hoses from the pressure control valve (on the forward side of the firewall).

(b) Disconnect the electrical leads from the valve at the electrical connector.

(c) Remove the screws and washers that secure the valve to the firewall and remove the valve.

(2) Installation

(a) Position the valve in place on the firewall and secure it with the previously removed screws and washers.

(b) Reconnect the two leads to the valve.

(c) Reconnect the two hoses to the valve.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING PNEUMATIC DEICE SYSTEM

Trouble	Cause	Remedy
Deicers do not inflate. Engine operating at minimum cruise RPM.	Open circuit breaker.	Push circuit breaker to reset.
	System connection loose or wire broken.	Tighten or repair as required.
	Timer not functioning.	Test or replace as required.
	Pressure regulator valves not functioning.	Make electrical test. Check for sticking poppet. Clean. Ensure that both vent ports on solenoid valves are open.
	Lines blocked or not connected.	Blow out lines and inspect connections. Make air leakage test.
Deicers do not inflate or inflate slowly.	Pressure regulator valves clogged.	Remove and clean.
Deicers inflate slowly (normal time - 6 seconds).	Lines partially blocked or not connected securely.	Blow out lines and inspect connections. Make air leakage test.
	Deflate solenoid valves not functioning properly.	Ensure that both vent ports on solenoid are open.
	System pressure not being reached.	Check performance to manufacturer's specifications.
	Deicer puncture.	Repair per specification or replace.
Deicers deflate slowly.	Lines partially blocked.	Inspect and blow out lines.
	Deflate solenoid valve not functioning properly.	Ensure that both vent ports on solenoid are open.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING PNEUMATIC DEICE SYSTEM

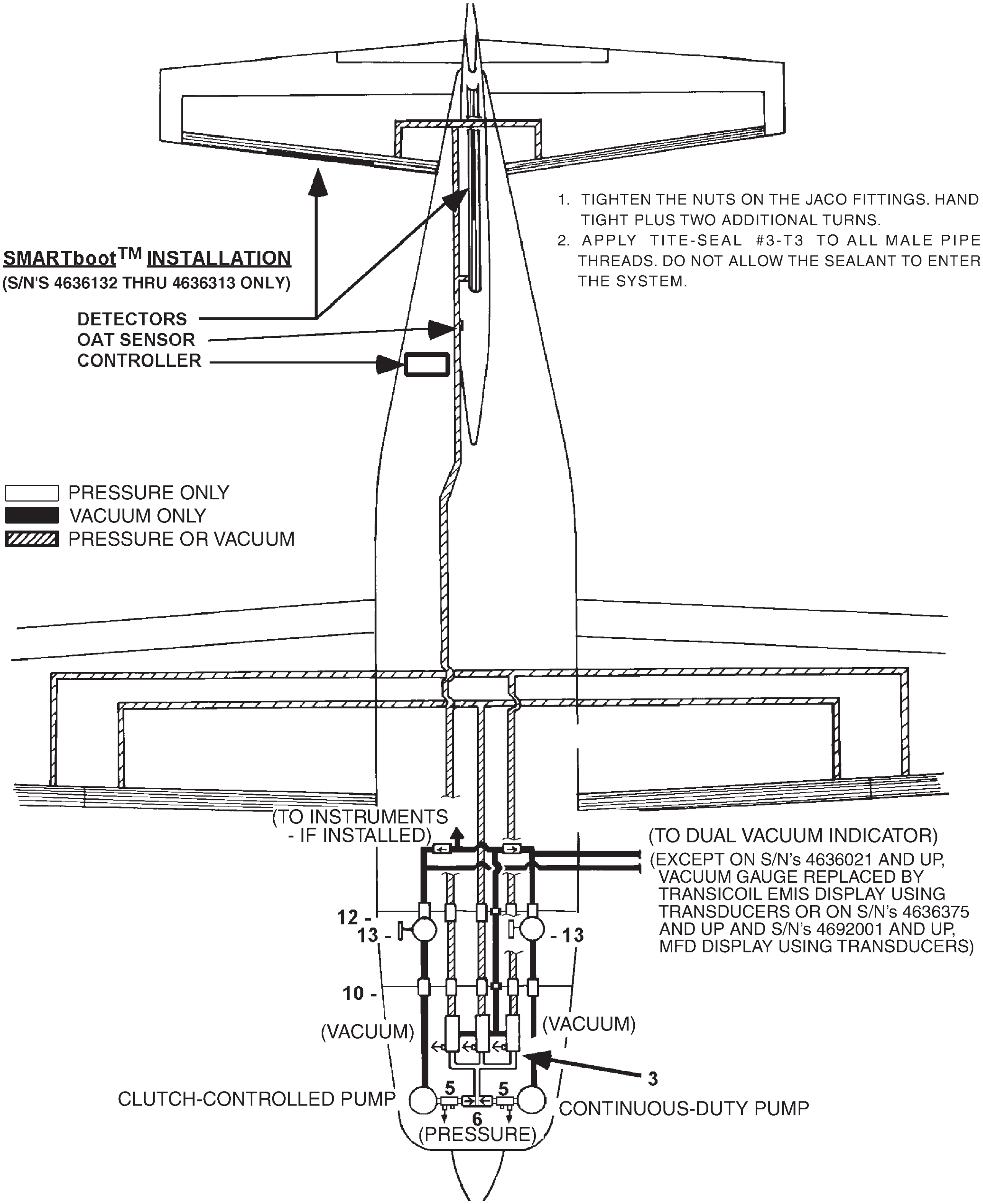
Trouble	Cause	Remedy
Deicers inflate, indicator light does not function. (Check that deicer boot switch is ON.)	Indicator lamp burned out.	Replace lamp.
	System pressure not being reached.	Check "Deicers Inflate Slowly" above.
	Pressure regulator valve switch not functioning.	Make electrical test and replace if required.
	Wires loose or broken.	Make electrical test. Repair or replace broken wires.
	Poor grounding of pressure regulator valve switch.	Check for proper ground.
Deicer boots do not hold their form in flight or vacuum to the system is inadequate.	Deflate valve not functioning properly	Remove and trouble-shoot valve. Replace if necessary.
	Vacuum line restricted.	Disconnect line from instruments, deflate valve, and blow out line.
	Broken line.	Inspect system and repair.

D. Vacuum Manifold

The vacuum manifold (if installed) is located forward of the forward pressure bulkhead, slightly to the left of the airplane centerline.

(1) Removal
- (a) Disconnect the three (or four) hoses from the manifold body.
- (b) Remove the screw and washer which secures the manifold bracket to the pressure bulkhead.
- (c) Remove the vacuum manifold.

(2) Installation
- (a) Position the vacuum manifold in place.
- (b) Secure the vacuum manifold in place with the previously removed washer and screw.
- (c) Reconnect the three (or four) hoses to the manifold body.



Fluid Diagram of Pneumatic Deicer System (Typical)
Figure 8 (Sheet 1 of 2)

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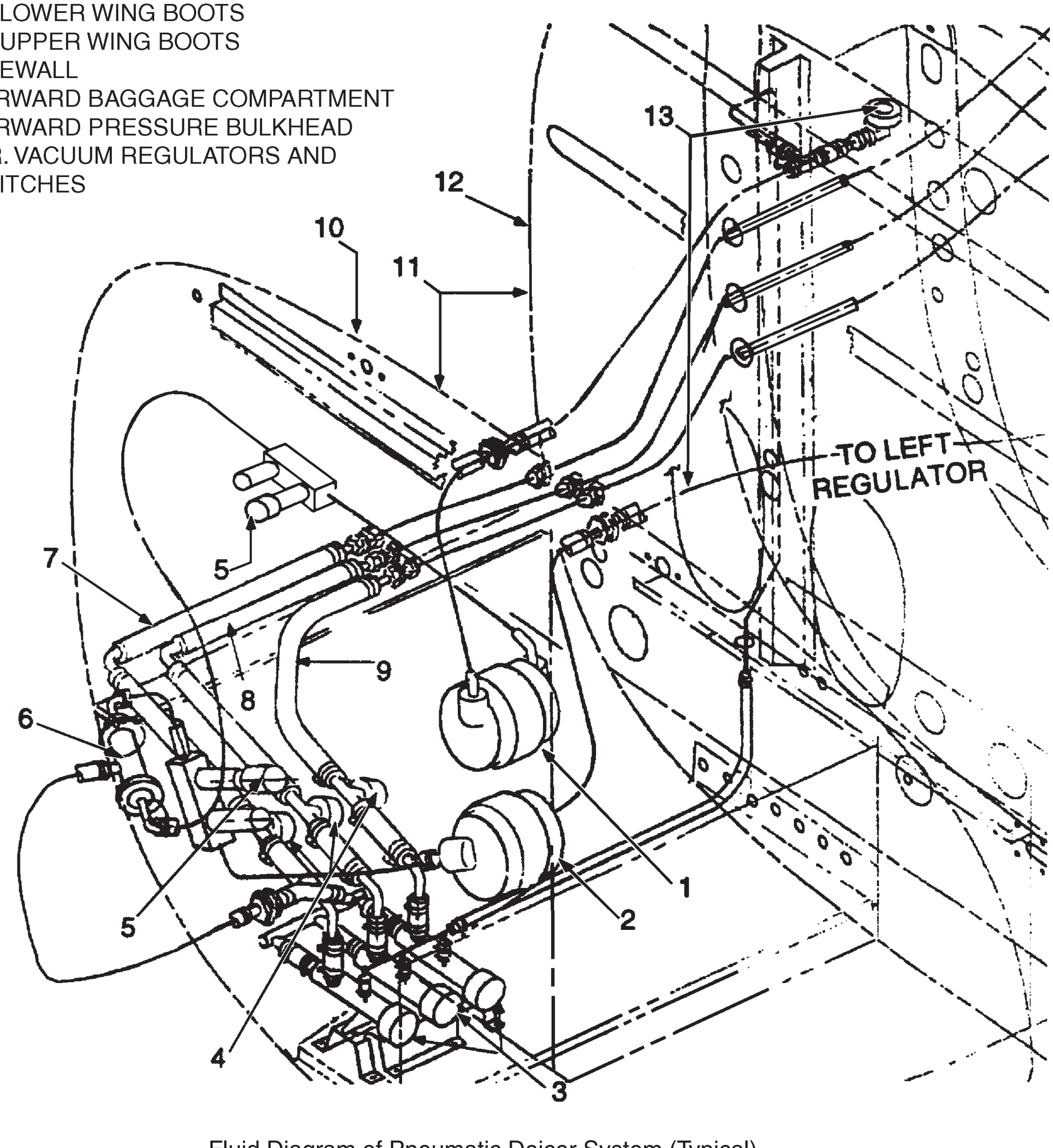
1. UPPER VACUUM PUMP (ON BACK SIDE OF ENGINE)
2. LOWER VACUUM PUMP (ON BACK SIDE OF ENGINE)
3. FLOW VALVE ASSEMBLY
4. PRESSURE SWITCHES
5. PNEUMATIC PRESSURE CONTROL VALVES
6. PNEUMATIC PRESSURE MANIFOLD / CHECK VALVE ASSEMBLY
7. TO EMPENNAGE BOOTS
8. TO LOWER WING BOOTS
9. TO UPPER WING BOOTS
10. FIREWALL
11. FORWARD BAGGAGE COMPARTMENT
12. FORWARD PRESSURE BULKHEAD
13. L. R. VACUUM REGULATORS AND SWITCHES

NOTE

ITEMS 1 AND 2 LOCATED ON REAR RIGHT SIDE OF ENGINE. ITEMS 3 AND 4 LOCATED UNDER FLOOR IN BAGGAGE COMPARTMENT. ITEM 5 AND 6 LOCATED ON ENGINE (RIGHT) SIDE OF FIREWALL. ITEM 13 LOCATED ON BAGGAGE COMPARTMENT SIDE OF PRESSURE BULKHEAD.



Fluid Diagram of Pneumatic Deicer System (Typical)
Figure 1 (Sheet 2 of 2)

E. Pressure Manifold/Check Valve

The pressure manifold/check valve is located on the forward side of the firewall to the right.

(1) Removal

(a) Disconnect the three hoses from the pressure manifold/check valve.

(b) Remove the screws that secure the pressure manifold/check valve to the firewall.

(c) Remove the pressure manifold/check valve.

(2) Installation

(a) Position the pressure manifold/check valve on the forward side of the firewall. Secure it with the screws.

(b) Reconnect the three hoses to the pressure manifold/check valve.

(3) Testing

(a) Remove pressure manifold/check valve by disconnecting vacuum pump discharge lines and the pressure line to the pressure control valve.

(b) Connect a line from a regulated air source to the pressure control manifold at the pressure control valve connection.

(c) Slowly apply 20 psig of air pressure to the pressure manifold/check valve.

(d) Close off regulated air.

(e) Check for leaks by noting the pressure drop in the pressure manifold/check valve. A pressure drop of not more than 5 psig in 10 seconds is permitted.

(f) If air flows through the pressure manifold/check valve and out the vacuum pump discharge side of the manifold, the diaphragm is ruptured and the manifold must be replaced.

(g) If unit passes leak test, or if a new manifold is required, install in the airplane by connecting both vacuum pump discharge and the pressure control valve hoses.

F. Timer

The pneumatic system timer is mounted to the upper right hand corner of the forward side of the F.S. 108.00 frame.

(1) Removal

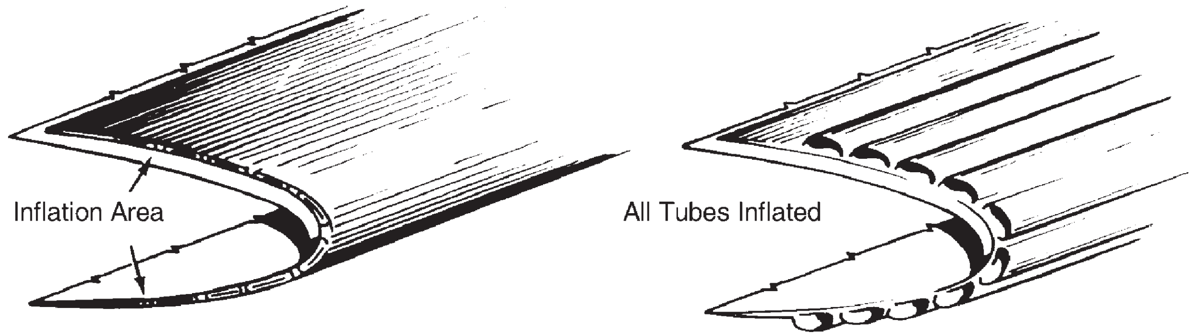
(a) Disconnect the electrical leads from the timer at the connector.

(b) Remove the screws, nuts and washers that secure the timer to the frame. Remove the timer.

(2) Installation

(a) Position the timer on the F.S. 108.00 frame. Secure it in place with the screws, nuts and washers.

(b) Reconnect the electrical leads at the connector.



Pneumatic Deicer Boot Operation
Figure 9

4. Deicers (Boots) (Pneumatic Deice System)

Boots are attached to the leading edge of the wings, vertical stabilizer, and elevator. The boots are made up of a fabric-reinforced rubber containing built-in span-wise inflation tubes. The boots are secured to the leading edges with cement and are connected to the plumbing through the skin by flexible or aluminum air connections.

WARNING: CEMENTS AND SOLVENTS USED TO REMOVE AND INSTALL DEICERS ARE EXTREMELY FLAMMABLE AND TOXIC. EXTINGUISH OPEN FLAMES. AVOID SPARKS. USE IN WELL-VENTILATED AREA. AVOID SKIN CONTACT AND/OR PROLONGED BREATHING OF VAPORS. CONSULT MSDS FOR ADDITIONAL SAFETY INFO.

CAUTION: DISPOSE OF UNUSED MEK AND OTHER CHEMICALS AND SOLVENTS IN A MANNER CONSISTENT WITH LOCAL LAWS AND/OR ENVIRONMENTAL PROTECTION AGENCY REGULATIONS.

CAUTION: DO NOT USE MEK ON SMARTBOOT™ INSTALLATIONS (VERTICAL FIN AND RIGHT STABILIZER) AS MEK WILL ATTACK ESTANE® MATERIAL. USE TOLUENE INSTEAD.

NOTE: Goodrich Black Standard Pneumatic De-Icer Installation, Maintenance & Repair Manual, ATA Report No. 30-10-31, provides approved, alternate procedures for removing and installing deice boots. Reference to it is highly recommended. It can be obtained online at http://www.goodrich.com/TechPubs.

A. Removal

WARNING: THE EXACT LOCATION OF WING STALL STRIPS IS CRITICAL TO THE AIRPLANE'S FLYING CHARACTERISTICS. THEREFORE, PRIOR TO REMOVING DEICER BOOTS, DETERMINE THE EXACT LOCATION OF THE STALL STRIPS. FABRICATE A TEMPLATE, OR DETERMINE EXACT VERTICAL AND HORIZONTAL MEASUREMENTS, TO ENSURE THE STALL STRIPS ARE INSTALLED IN THE SAME LOCATION AFTER REINSTALLING THE DEICER BOOTS.

See "Chart 2" on page 30108 for required materials.

NOTE: Disconnect line fittings from boot fittings.

(1) Starting at one corner of the upper trailing edge of the deicer, apply a minimum amount of solvent to the seam line while tension is applied to peel back the corner of the deicer.

CHART 2 (Sheet 1 of 2)
REQUIRED MATERIALS - PNEUMATIC DEICER - INSTALLATION / REMOVAL / REPAIR

Adhesive * - Choose One System (Goodrich Part Number)

3M 1300L System (preferred adhesive) 1300L Cement [Piper P/N 179-929 = 1 qt.]	and,	A-851-B Fuel Barrier Adhesive (74-451-99) = 1 qt.
or		
Bostik 1096M System 1007 or 1007M Bostik Primer 1096M Bostik Cement Boscodur 9R Accelerator	or,	British Bostik 2402 System 9252 British Bostik Primer 2402 British Bostik Cement Bostikure D Accelerator

* use Boar's Bristle brush

Filler - Choose One Product (If Required)

PRC1422 or PRC1425 or equivalent

Sealer * - Choose One Product (Goodrich Part Number)

A-56-B*** (74-451-11) = 1/2 pt **; (74-451-11-1) = 1 qt.	or,	Edge Sealer Kit (74-451-P) (Black Estane® or Neoprene Deicers) Edge Sealer Kit (74-451-K) (Black Estane® Deicers only)

* use Pure Bristle brush ** Piper P/N 279-142 *** Neoprene deicers only

Solvents

CAUTION: DO NOT USE MEK ON ESTANE® (I.E. - SMARTBOOTS™) DEICERS INTENDED FOR REUSE OR CONTINUED USE.

Item	Where Used
Toluene	Leading edge and deicer cleaning solvent
Toluene, or	Leading edge stripping solvent
Peerco #321 stripper, or	
Methyl Ethyl Ketone (MEK)	
MEK or Toluene or Acetone	Prop deicer removal and tackifying solvent *

* Toluene is the preferred tackifying solvent. Methylethylketone (MEK) can be used, but causes very rapid drying and provides only 10 seconds working time compared with 40 seconds for Toluene.

Tools and Miscellaneous (Goodrich Part Number)

2 x 2 1/2 in. Rubber roller (74-451-74)	1/4 in. Steel stitcher roller (74-451-89)
1 in. Paint brush	Lint-free cloths
Scissors	1 in. masking tape
Emery Buffing Stick (74-451-75) 6 ea.	Sharp knife
Steel measuring tape (6 ft.)	Fine sharpening stone
Steel wool pads	Hypodermic needles (22 gauge or smaller)
Straight edge	Carpenter's chaulk line
Hook knife (74-451-45)	3M EC 801 filler
FASTpatchTM Primer - 1/2 oz. crush tube (74-451-190)	

CHART 2 (Sheet 2 of 2)
REQUIRED MATERIALS - PNEUMATIC DEICER - INSTALLATION / REMOVAL / REPAIR

Goodrich Part No.	Goodrich Kits Quantity	Description
74-451-AA	**1**	**Universal FASTpatch™ Repair Kit**
74-451-206	1	Primer Pen (2 oz.)
74-451-187	30	Small Oval Patch (1-1/4 x 2-1/2 in.)
74-451-188	30	Medium Oval Patch (2-1/2 x 5 in.)
74-451-189	10	Large Oval Patch (5 x 10 in.)
74-451-75	3	Emery Cloth (9 x 11 in.)
74-451-87	1	Buffing Shield
74-451-AE	**1**	**Pinhole Repair Kit**
74-451-201	8.5 grams	Accelerator
74-451-202	1	Buffing Shield
74-451-205	1	Application Tool
74-451-209	24	Cement
74-451-210	24	Mixing Cup
74-451-211	24	Mixing Tool
74-451-212	50	Application Template
74-451-K	**1**	**Edge Sealer Kit (Black Estane® Deicers only)**
74-451-117-1	1 qt.	Sealer/Cement
74-451-120	4 oz.	Accelerator
74-451-L	**1**	**Resurfacing Kit ***
74-451-120	4 oz.	Accelerator
74-451-122	1 qt.	Primer
74-451-123	1 qt.	Resurfacing Coat
* after P/N 74-451-L is applied to a neoprene deicer, the deicer should be treated as an Estane® deicer for maintenance purposes.		
74-451-P	**1**	**Edge Sealer Kit (Black Estane® or Neoprene Deicers)**
74-451-20	1 pt.	Cement
74-451-148	50 ft.	Edge sealer strip
74-451-221	1 oz.	Accelerator
74-451-185	1 pt.	Primer
74-451-Z	**1**	**Maintenance Kit**
74-451-127	1 qt.	AgeMaster® No. 1
74-451-178	1 pt.	ShineMaster®
74-451-179	1 qt.	ShineMaster® Prep Cleaner
ICEX® II	16 oz.	Ice Adhesion Inhibitor

(2) Using a pressure handle squirt can filled with solvent, separate the deicer boot from the surface for a distance of four (4) inches all the way along the upper trailing edge.

(3) The area between the deicer and the wing which has now been separated will act as a reservoir for the solvent, therefore, the deicer can be pulled down towards the leading edge with a uniform tension.

(4) From the centerline of the leading edge to the lower trailing edge of the deicer, use the pressure handle squirt can to soften the bond between the deicer and the wing skin.

(5) Use Toluene or Peerco #321 to clean the dry cement off the exposed wing area, and clean the area thoroughly with MEK (methylethylketone).

B. Installation

(PIR-PPS50052, Rev. AB.)

See “Chart 2” on page 30108, for required materials.

(1) Leading Edge Preparation.

(a) Remove Lift Transducer (see 27-30-00) prior to fitting deicer on leading edge.

(b) Dry fit deicer on leading edge. Make sure deicer air connection fits in air connection holes in leading edge and cut-outs are properly aligned. Use deicer as template to mask installation area with one (1) inch masking tape. For non-recessed deicers, add at least one-half (1/2) inch extra around perimeter of deicer. Mark deicer centerline on masking tape at each end.

NOTE: Masking accurately will eliminate the need for cleaning off excess cement later.

(c) Remove paint and primer in masked area with stripper or sand painted/primed area to roughen. Use clean pressurized air or dry cloth to remove sanding particles before proceeding with deicer installation.

NOTE: Deicers can be installed on alodined or anodized surfaces, and over zinc chromate primer that cannot be removed by scrubbing with solvent.

(d) Clean the surfaces thoroughly, at least twice, with acetone. For final cleaning, wipe the solvent film off quickly with a clean dry cloth before it has time to dry.

(e) Fill gaps of skin splices that lead under deicers with “filler” from “Chart 2” on page 30108.

(f) On wings only, apply fuel barrier cement over fuel tank rivets and edges of fuel tanks which will lie under the installed deicer. Proceed as follows:

1) Stir cement thoroughly to blend solids.

2) Apply one even brush coat and let dry 1/2 hour (30 minutes).

3) Apply a second coat and let dry two (2) hours.

(g) Remove the sump plugs from the air connection grommets. In some cases, it will be necessary to remove sections of doped fabric used to cover the air connection holes. Draw out the ends of the non-kink hose section so that they protrude through the connection holes in the leading edge. If hose is cracked or deteriorated, replace with new hose.

(2) Deicer Preparation.

CAUTION: DO NOT USE MEK TO CLEAN SMARTBOOTS™ OR ESTANE® AS MEK WILL ATTACK ESTANE® MATERIAL.

(a) Moisten a clean cloth with Toluene and carefully clean the rough, back surface of the boot at least twice.

(b) Change cloths often to avoid recontaminating the cleaned area.

(3) Cement Application.

CAUTION: DO NOT INSTALL IF HUMIDITY IS ABOVE 90% OR IF TEMPERATURE IS BELOW 50° F. AT THESE EXTREMES, PROPER ADHESION MAY NOT BE OBTAINED REGARDLESS OF DRYING TIME ALLOWED.

CAUTION: DRY FIT DEICER BEFORE APPLICATION OF CEMENT. SEE LEADING EDGE PREPARATION, ABOVE.

NOTE: Deicer and leading edge may be cemented for a maximum of 48 hours before actual installation, if cemented parts are covered and kept clean.

(a) Thoroughly mix cement before using. Apply one even brush coat to the cleaned back surface of the boot and to the cleaned metal surface of the leading edge.

(b) Allow the cement to air dry for a minimum of one hour at humidity of 75% or lower. If humidity is 75-90%, drying time will be longer.

(c) Snap a chalk line along the leading edge of the airfoil section. Intensify chalk line on leading edge and the white reference line on the boot with a ball point pen.

(d) Stir cement and apply a second coat to both surfaces and allow to air dry per (b), above.

(4) Mounting Deicer on Leading Edge.

CAUTION: DO NOT TRIM DEICER CLOSER THAN ONE (1) INCH FROM TUBE AREA AT ENDS OR CUT-OUT AREAS, AS AIR LEAKAGE OR SEAM SEPARATION CAN RESULT.

Most boots are made with an excess of material at the inboard and outboard edges for final trimming after installation and some recessed boots trim on the upper and lower edges.

(a) Holding the backside of the boot close to the leading edge, fasten the end of each non-kink hose to the corresponding air connection stem. Tinnerman or other suitable non-kink hose clamps should be used for this purpose. Tighten each clamp with a pair of slip joint pliers but do not squeeze the clamp so tight that the hose is damaged.

NOTE: If non-kink hose clamps are not available, wrap each hose connection with several turns of friction tape. Over the tape wrap two separate bindings of safety wire, about one-half (1/2) inch apart. Each of these bindings should consist of several turns of wire. Twist together the ends of each binding to tighten. Press the twisted ends down against the hose. Finally, wrap the wire with several additional turns of friction tape.

(b) Push the hose connections into the leading edge grommets or seals, as the case may be. Obtain sufficient personnel to hold boot steady during installation. (Limit handling cemented side of boot with fingers.)

(c) Continue installation by reactivating the cement along the centerline leading edge surface and boot in spanwise strips approximately three (3) to four (4) inches wide by eighteen (18) to thirty-six (36) inches long. Use a clean lint free cloth dampened with Toulene. Wring and shake cloth to remove excess solvent.

NOTE: Application of vacuum is recommended and makes installation easier.

(d) Using a rubber roller, press the deicer firmly against the wing leading edge, being careful not to trap any air under the deicer. Always roll parallel to the inflatable tubes. Position the deicer centerline to coincide with leading edge centerline. Hold boot in this position while reactivating about three (3) inches around connections and around corresponding holes in leading edge, using a clean, lint-free cloth moistened with Toluene. Insert connections in leading edge holes when cement has dried to a tacky state, and press boot to leading edge with roller in tackified area.

(e) If the deicer should attach “off course,” use MEK to remove and reposition properly. Avoid twisting or sharp bending of the deicer.

(f) Using the rubber roller, apply pressure over entire surface of the deicer. All rolling should be done parallel to the inflatable tubes. Roll edges with a one-quarter (1/4) inch stitcher roller.

CAUTION: AVOID EXCESSIVE SOAKING OR RUBBING OF THE CEMENT WHICH COULD REMOVE THE CEMENT FROM THE SURFACE.

Remove all masking tape and clean surfaces carefully with Toluene so that no solvent will run under deicer edges.

(g) Reinstall Lift Transducer (see 27-30-00).

Bewteen Stall Warning Transducer gasket and De-Ice Boot fill air gap all around using Type 8 sealant, see 91-10-00 Chart 11, Consumables - Sealants.

(h) Mask areas to be edge filled - i.e. - fair full thickness deicer edges to adjacent aircraft structure, around cut-outs, recessed edges, and/or between two deicers installed close together. Apply two lines of masking tape bordering fill area:

1) Apply one line on deicer surface one-sixteenth (1/16) to one-eighth (1/8) inch from edge; and,

2) One line on leading edge one-quarter (1/4) to one-half (1/2) inch from deicer edge.

NOTE: Width between masking tape lines may vary depending on area to be filled; use narrowest width possible.

(i) Apply edge filler to fair full thickness deicer edges to adjacent aircraft structure, around cut-outs, recessed edges, and/or between two deicers installed close together.

(j) Let edge filler dry per manufacturer’s instructions.

(k) Apply edge sealer using one of the following methods, appropriate to the deicer installed:

1) A-56-B (Neoprene deicers only.)

a) Apply masking tape on deicer surface around deicer and cutouts at least 1/4 inch from edge. If edge filler has been used, make sure masking tape applied on deicer is at least 1/16-1/4 inch in from edge filler.

b) Apply masking tape on leading edge one-eighth (1/8) to one-quarter (1/4) inch outboard of area initially cleaned and cemented (about 3/4 inch out from deicer edge). Form neat, straight lines to border edge sealer application.

c) Mix edge sealer (A-56-B cement) thoroughly. If desired, dilute with Toluene, not to exceed 20% by volume.

d) Apply an even brush coat of edge sealer to surfaces between tape lines. Ensure that the conductive edge sealer coating is continuous from the deicer surface to the wing painted surface.

e) Remove tape immediately after applying edge sealer - i.e. - before edge sealer dries.

2) Edge Sealer Kit No. 74-451-P (See “Figure 10”.)

a) Apply masking tape on deicer surface around deicer and cutouts at least 1/4 inch from edge.

b) Apply masking tape on leading edge at least one and three-quarters (1 3/4) inch outboard of edge of masking tape applied to deicer and outboard of exposed 1300L cement. Form neat, straight lines to border edge sealer application.

c) For neoprene deicers, brush one even coat of primer on leading edge and deicer between tape lines.

d) For Estane® deicers, brush one even coat of primer on leading edge surface only, between the deicer edge and the masking tape. Do not get primer on deicer surface. Let dry thirty (30) minutes.

e) Mix cement and accelerator per instructions on container.

f) Apply one even brush coat of edge sealer to surfaces between tape lines. Let dry fifteen (15) minutes.

CAUTION: EDGE SEALER STRIP HAS A PEEL PLY ON BREEZE SIDE. REMOVE PEEL PLY A LITTLE AT A TIME AS YOU INSTALL STRIP AS STRIP MAY STICK TO ITSELF ON CONTACT.

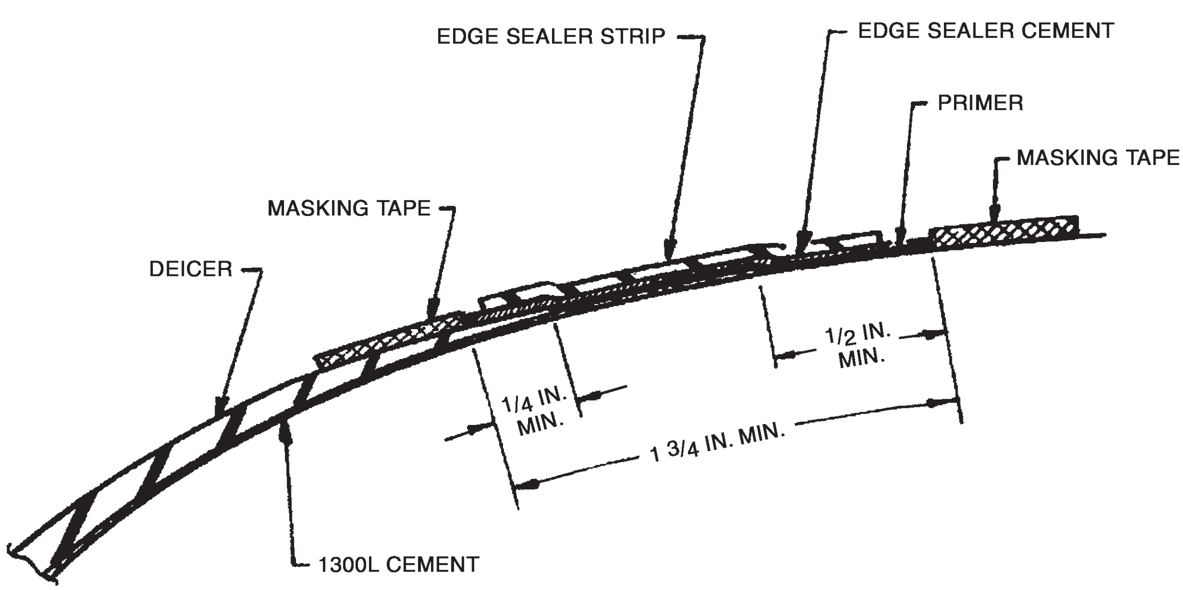
CAUTION: DO NOT ALLOW MEK TO COME CONTACT SURFACES OTHER THAN BACKSIDE OF EDGE SEALER STRIP AS IT WILL ATTACK ESTANE® DEICER MATERIAL. DO NOT SATURATE EDGE SEALER STRIP DURING TACKIFICATION FOR SAME REASON.

g) Beginning on the upper spanwise edge of deicer, tackify backside of strip (side without peel ply) with MEK in a fifteen (15) to twenty (20) inch length and press strip onto cemented area between tape lines. Make sure strip contacts deicer surface and leading edge surface. Remove peel ply on breeze side of strip and roll with rubber roller. Continue until one side of deicer is edged.

h) At corners, apply strip about one (1) to one and one-half (1 1/2) inches beyond deicer end.

i) Repeat Steps g & h for lower edge of deicer.

j) Repeat Steps g & h for chordwise edges of deicer, overlapping strip on top of spanwise strips. This arrangement provides aerodynamic flow to minimize drag and possibility of strip lifting.



Edge Sealer Kit P/N 74-451-P Application
Figure 10

3) Edge Sealer Kit No. 74-451-K (Estane® deicers only.)

a) Apply masking tape on deicer surface around deicer and cutouts at least 1/4 inch from edge. If edge filler has been used, make sure masking tape applied on deicer is at least 1/16-1/4 inch in from edge filler.

b) Apply masking tape on leading edge one-eighth (1/8) to one-quarter (1/4) inch outboard of area initially cleaned and cemented (about 3/4 inch out from deicer edge). Form neat, straight lines to border edge sealer application.

c) Mix only the amount of edge sealer that can be used in a four (4) hour period. Mix cement and accelerator per instructions on container.

d) Apply one even brush coat of edge sealer to surfaces between tape lines. Ensure that the conductive edge sealer coating is continuous from the deicer surface to the wing painted surface.

e) Remove tape immediately after applying edge sealer - i.e. - before edge sealer dries.

(5) Drying Time.

(a) Allow a minimum of four (4) hours drying time before flying airplane.

(b) Allow a minimum of 48 hours drying time before inflating deicers.

(6) Bonding Stall Strips

(a) Surface Preparation

1) Using the template created when removing stall strips, mask off area to be covered by stall strip.

2) Remove gloss from boot in masked off area by sanding lightly using 320 grit or finer abrasive paper or ScothBrite Ultrafine Pads.

3) Remove gloss from curved surface at stall strip by sanding lightly using 320 grit or finer abrasive, or ScothBrite Ultrafine Pads.

4) Clean bonding surface of boot and stall strip using clean lint free cloth moistened with toluene, naphtha, or MEK.

5) Clean surface again, but this time wipe the solvent film off quickly before it dries using a clean dry cloth.

(b) Bond stall strips using 3M adhesive EC-1300L or 1300L, then fill and feather edges using a Type 8 sealant, such as Flamemaster CS3247 B-4, CS3247 B-2, or PPG PR 1425CF B-2.

(7) Adhesion Test.

(a) Using excess boot material trimmed from the ends of any wing and empennage deicers, prepare one test specimen for each deicer installed. This specimen should be a one (1) x four (4) inch full thickness strip of boot material cemented to the wing skin adjacent to installed boot following the identical procedure used for installation. Leave one inch of the strip uncemented to attach a clamp. Four (4) hours or more after the installation, attach a spring scale to the un-cemented end of each strip and measure the force required to remove strip at the rate of one inch per minute. The pull should be applied 180° to the surface. (Strip doubled back on itself.)

CAUTION: SUCCESSFUL COMPLETION OF THIS TEST INDICATES INSTALLATION IS SUITABLE FOR FLIGHT, NOT INFLATION.

(b) A minimum of five pounds tension (pull) shall be required to remove the test strip. If less than five pounds is required, wait two to three hours and retest. If peel strength is still lower than five pounds, proceed as follows:

1) Carefully lift one corner of deicer in question sufficiently to attach a spring clamp.

2) Attach a spring scale to this clamp and pull with force 180° to the surface and in such a direction that the boot tends to be removed on the diagonal.

3) If a force of five pounds per inch of width can be exerted under these conditions, the installation shall be considered satisfactory. Remember, the width increases as the corner peels back.

4) Re-cement corner following previous procedure.

5) Failure to meet this requirement shall require reinstallation of the deicer.

NOTE: Possible reasons for failure are: dirty surfaces, cement not reactivated properly, cement not mixed thoroughly. Corrosion of the metal skin may occur if good adhesion is not attained, especially around rivet heads and metal skin splices.

C. Inspection

Perform the following inspection when:

(1) For an uninstalled deicer:

(a) The deicer has been stored for more than 84 months.

(b) There is evidence of damage to original packaging.

(c) The deicer is not in original packaging.

(d) There is evidence that deicer was not stored properly.

(2) For an installed deicer if damage or age is suspected of affecting deicer performance.

(3) Procedure

CAUTION: INSTALLATION OR CONTINUED USE OF A DEICER IN CONDITIONS SPECIFIED IN (1) OR (2), ABOVE, IS AT USER'S DISCRETION.

NOTE: Check for damage with deicer inflated using leak detector fluid. Air leaking damage can be located and marked for repair. Deicer should not be inflated at pressures higher than indicated in "Chart 3".

(a) Inspect deicer carefully for surface damage: cuts, tears, abrasions, scuffs, cracking and/or crazing. Check backside (if uninstalled) and breeze side of deicer carefully. Pay particular attention to air connection area.

CAUTION: REVIEW REPAIR LIMITS, BELOW, TO DETERMINE IF REPAIR IS PERMITTED.

(b) Repair surface damage detected per "Repair Procedures" on page 301019.

(c) Inflate deicer with regulated air source to correct operating pressure (see "Chart 3"). Check inflation time. Deicer should inflate to operating pressure within six (6) seconds.

CHART 3
PNEUMATIC BOOT OPERATING PRESSURES

Deicer	Operating Pressure (PSI)	Inflation Sequence
Wing	14 - 15	Alternating (Upper and Lower tubes inflate separately)
Vertical Fin	10 - 12	Simultaneous
Horizontal Stabilizer	10 - 12	Simultaneous

(d) When deicer has reached operating pressure, seal off deicer at air connection. Check deicer pressure after 60 seconds. Pressure drop should not exceed three (3) psi for deicers with system operating pressure of 14 psi or higher.

(e) Allow deicer to deflate naturally with no vacuum applied. Deflation time should not exceed 22 seconds. When deicer is deflated, check for pockets of trapped air in tubes.

CAUTION: REVIEW REPAIR LIMITS, BELOW, TO DETERMINE IF REPAIR IS PERMITTED.

(f) If deicer does not pass these tests, check again for damage, perform appropriate repairs and retest. If deicer still does not meet test criteria, the deicer should be replaced (if installed) or scrapped (if uninstalled).

(g) If deicer passes all tests, its usability is on condition and the decision to install is at user's discretion.

D. Maintenance

CAUTION: AVOID BOOT EDGE SEALER/FILLER WHEN APPLYING ANY PRESERVATIVE / COSMETIC COATING OR ICE INHIBITOR TO DEICE BOOTS. SUCH COATINGS MAY LESSEN THE ADHESIVE PROPERTIES OF THE EDGE SEALER/FILLER.

(1) Visually inspect deicers frequently for abrasion, erosion, cracking, tears, pinholes, foreign object damage (FOD), debonding and other damage. Repair damage immediately, see Repairs, below.

(2) Clean deicers when the airplane is washed with a mild soap and water solution. In cold weather, wash the boots with the airplane inside a warm hangar if possible. If the cleaning is to be done outdoors, heat the soap and water solution before taking it out to the airplane. If difficulty is encountered with the water freezing on the boots, direct a blast of warm air along the region being cleaned, using a portable type ground heater.

(3) Goodrich Corp. offers a family of deicer treatment products which reduce weathering effects, enhance appearance, and enhance performance. Their use is recommended, see below.

(a) AGEMASTER® NO. 1 Application.

AgeMaster® No. 1 is a rubber preservative for neoprene deicers only that protects against weathering, ozone and ultraviolet rays. Apply AgeMaster® No. 1 initially 6 months after a new deicer is installed. Reapply every 6 months.

WARNING: AGEMASTER® NO. 1 CONTAINS PETROLEUM DISTILLATES. HARMFUL OR FATAL IF SWALLOWED. IF SWALLOWED, DO NOT INDUCE VOMITING; SEE PHYSICIAN IMMEDIATELY. KEEP AWAY FROM OPEN FLAME. VAPORS MAY IGNITE CAUSING FLASH FIRE OR EXPLOSION. DO NOT APPLY BY SPRAYING. USE WITH ADEQUATE VENTILATION. AVOID PROLONGED BREATHING OF VAPOR. IF DIZZINESS OR NAUSEA OCCURS, OBTAIN FRESH AIR. AVOID CONTACT WITH SKIN AND EYES. IF EYE CONTACT OCCURS, FLUSH EYES WITH WATER FOR 15 MINUTES, THEN SEE PHYSICIAN. IF SKIN CONTACT OCCURS, WASH THOROUGHLY WITH SOAP AND WATER. EMPTY CONTAINERS MAY CONTAIN FLAMMABLE OR EXPLOSIVE RESIDUAL VAPORS. SEE MSDS FOR ADDITIONAL SAFETY INFORMATION.

CAUTION: AGEMASTER® NO. 1 STAINS SKIN, CLOTHING AND OTHER SURFACES. WEAR PLASTIC OR RUBBER GLOVES WHEN USING. PROTECT SURROUNDING AREAS. USE WATERLESS HAND CLEANER TO REMOVE STAINING.

CAUTION: NOT SUITABLE FOR USE ON ESTANE® DEICERS. IT WILL NOT BE ABSORBED BY DEICER, CAUSING RUN BACK AND STAINING ON AIRCRAFT SURFACE.

NOTE: AgeMaster® No. 1 can be applied to neoprene deicers that have been repaired with Goodrich P/N 74-451-AA patch kit and/or Goodrich P/N 74-451-AE pinhole repair kit; however, excess AgeMaster® No. 1 should be wiped off the surfaces of the repair material, as it will not be absorbed.

NOTE: Store in cool, well-ventilated place. Keep container closed tightly when not in use.

1) One quart covers 90 square feet of deicer, following these directions.

   Thoroughly clean deicer surface with mild soap and water. Rinse with clean water and let dry.

2) Use isopropyl alcohol to remove substances that cannot be removed with soap and water. Repeat Step 1, above.

3) Wipe one even coat of AgeMaster® No. 1 on deicer surface with lint free cloth. Coat deicer surface completely and evenly for best results and appearance. Let dry 5-10 minutes. Dry time may vary due to temperature and humidity conditions.

4) Repeat Step 3, above, so that three even coats have been applied with 5-10 minutes dry time between each coat.

5) Let dry 24 hours before flying aircraft or applying Icex® II or ShineMaster® products.

(b) ShineMaster® Application.

ShineMaster® treatment is a cosmetic coating that provides high luster shine. ShineMaster® Prep cleaner is used to clean deicer for ShineMaster® application, and remove residual ShineMaster® before reapplication. Application interval for ShineMaster® depends on operating environment (2 or 3 times per year is typical).

NOTE: ShineMaster® products can be applied to deicers that have been repaired with Goodrich P/N 74-451-AA patch kit and/or Goodrich P/N 74-451-AE pinhole repair kit.

NOTE: ShineMaster® cosmetic treatment should not be heavily applied or allowed to build-up on deicer edges, as it may appear cracked and is very difficult to remove when reapplication is desired.

1) Clean deicer surface with ShineMaster® Prep cleaner to remove dirt, grease, oil, silicone products and other contamination. Previous applications of ShineMaster® treatment must be removed before reapplication to avoid dulling or product build-up.

2) Apply light, even coat of ShineMaster® cosmetic treatment to deicer with clean lint-free cloth wiping in one direction. Let dry to touch - about 5-15 minutes.

3) Repeat Step 2, above, once or twice to obtain desired shine.

(c) Icex® II Application.

Icex® II is an ice adhesion inhibitor that enhances deicer performance by lowering adhesion strength between ice and deicer surface. During icing season, apply Icex® II every 50 flight hours.

NOTE: Icex® II is not a cure-all for icing problems. Icex® will not prevent or remove ice formations. Its only function is to keep ice from initially getting a strong foothold, thus making removal easier.

NOTE: Icex® II can be applied to deicers that have been repaired with Goodrich P/N 74-451-AA patch kit and/or Goodrich P/N 74-451-AE pinhole repair kit.

NOTE: If Icex® II is applied too heavily, result can be sticky surface that collects dust and dirt, reducing efficiency of Icex® II. Residue should be completely removed before reapplication.

1) Clean deicer surface using mild soap and water.
2) Rinse with clean water and let dry.
3) Clean with isopropyl alcohol to remove substances not removed with soap and water.
4) Repeat Steps 1 and 2, above.
5) Apply Icex® II with clean cloth or pad. Apply lightly and wipe in single continuous back and forth motion spanwise on deicer.

(d) Application in Combination.

AgeMaster® No. 1, ShineMaster® and Icex® II are specifically designed to use together to provide maximum deicer care, appearance and performance.

1) Clean deicer surface to remove previous coatings, dirt, grease, oil and other contamination. ShineMaster® Prep cleaner should be used to remove ShineMaster® treatment.
2) Apply AgeMaster® No. 1, see above, and let dry at least 24 hours.
3) Apply ShineMaster® cosmetic treatment, see above, and let dry to touch.
4) Apply Icex® II, see above.
5) Icex® II may be removed with alcohol or warm, soapy water, then reapplied without affecting ShineMaster® cosmetic treatment.
6) Icex® II and/or ShineMaster® cosmetic treatment can be removed without affecting AgeMaster® No. 1.
7) Icex® II and/or ShineMaster® cosmetic treatment must be removed before reapplying AgeMaster® No. 1.

E. Repair Limits

CAUTION: PNEUMATIC DEICERS MUST BE REPLACED IF: (1) CUTS, TEARS OR RUPTURES CUT THE INFLATABLE TUBE FABRIC AND/OR EXCEED 4 X 9 INCHES; OR, (2) BROKEN STITCHES OR THREADS ARE APPARENT; OR, (3) DAMAGE LEAKS AIR AND EXCEEDS THE PATCH AND PINHOLE REPAIR LIMITS SPECIFIED BELOW.

NOTE: In addition to meeting these repair limits, the deicer must also pass the Inspection (Pneumatic Deicers (Boots)), Procedure, below.

NOTE: If patches and pinhole repairs are mixed within any 12 inch square, a ratio equivalent to the repair limits should be followed. For example, in a 12 inch square it would be permissible to have one (1) small patch (33% of repair limit) and 13 pinhole repairs (67% of repair limit). The net effect is equivalent to 100% of the repair limits.

To maintain optimal functional efficiency of the deicers:

(1) The following patch limits are established:

(a) Three (3) small patches per 12 inch square (Small patch is 2-1/2” X 1-1/4”); or,
(b) Two (2) medium patches per 12 inch square (Medium patch is 5” X 2-1/2”); or,
(c) One (1) large patch per 12 inch square (Large patch is 5” X 10”); or,
(d) Two (2) small patches and one (1) medium patch per 12 inch square.

(2) No more than 20 pinhole repairs are allowed per 12 inch square.

F. Repair Procedures

CAUTION: REVIEW REPAIR LIMITS, ABOVE, TO DETERMINE IF REPAIR IS PERMITTED.

See “Chart 2” on page 30108 for materials and supplies. Kits and part numbers referenced below are Goodrich Corporation, unless otherwise noted.

NOTE: Repair of damage that does not leak air is not mandatory.

(1) Scuff (Surface) Damage: This type of damage will be most commonly encountered and, fortunately, it is not necessary in most cases to make a repair. Surface damage that leaks air, but does not cut inflatable tube fabric, and does not exceed 4” X 9” should be repaired with P/N 74-451-AA patch kit as follows:

CAUTION: PATCH ADHESION IS TEMPERATURE SENSITIVE. IF TEMPERATURE IS UNDER 50°F (10°C), WARM DEICER SURFACE PRIOR TO APPLYING PRIMER, AND WARM INSTALLED PATCH WHILE DRYING. TO WARM, HOLD PLASTIC BAG FILLED WITH HOT WATER ON DEICER SURFACE, AND ON INSTALLED PATCH. IF SURFACE AND INSTALLED PATCH ARE NOT WARMED, PATCH MAY NOT ADHERE.

CAUTION: PATCHES HAVE A ONE-WAY STRETCH ACROSS WIDTH OF PATCH SO PATCH CAN STRETCH WHEN DEICER INFLATES. PATCH MUST BE INSTALLED WITH LENGTH PARALLEL TO DEICER TUBES. FAILURE TO DO SO MAY RESULT IN PATCH LIFTING WHEN DEICER INFLATES.

CAUTION: PREPARATION OF DEICER SURFACE TO BE PATCHED IS CRUCIAL FOR GOOD PATCH ADHESION. FAILURE TO REMOVE, SHINEMASTER® COSMETIC TREATMENT, WAX, GREASE, OIL, ICEX® II, AND OTHER CONTAMINANTS COMPLETELY, CAN RESULT IN POOR PATCH ADHESION.

(a) Trim standard patch sizes to accommodate small areas of damage, leaving at least 1/2 inch beyond damaged area. Mark stretch direction on patch before trimming, so trimmed patch is installed with stretch in same direction as tube inflation.

(b) Clean deicer surface to be patched with detergent and hot water, using lint free cloths to remove dirt, grease and cosmetic coatings. Repeat until oil and silicone contamination is removed.

(c) Use buffing shield or patch as template to outline damaged area. Buff deicer surface with medium grit emery cloth or equivalent.

(d) Wipe deicer surface with cloth dampened with Toluene or alcohol and let dry.

(e) Apply one coat of primer to deicer surface. Let dry to touch (5-10 min.).

(f) Remove paper backing from patch and press patch on primed deicer surface. Roll with rubber roller.

(g) Let dry 30 minutes before inflating deicer.

(h) Patch refurbishment:

1) To replace loose or damaged patch, remove old patch by peeling off deicer.

2) Remove adhesive remaining on deicer with Scotch Brite™ soaked in alcohol.

3) Install new patch.

(2) Tube Area Damage: Use Kit No. 74-451-AA (see application instructions, under Scuff Damage, above) to repair cuts,tears or ruptures that leak air but do not cut inflatable tube fabric, and do not exceed 4” X 9”. Use Kit No. 74-451-AE to repair pinholes that leak air, but do not exceed 1/16 inch in diameter, as follows:

CAUTION: CUTS, TEARS, OR RUPTURES THAT CUT THE INFLATABLE TUBE FABRIC REQUIRE DEICER REPLACEMENT.

CAUTION: PATCHES HAVE A ONE-WAY STRETCH ACROSS WIDTH OF PATCH SO PATCH CAN STRETCH WHEN DEICER INFLATES. PATCH MUST BE INSTALLED WITH LENGTH PARALLEL TO DEICER TUBES. FAILURE TO DO SO MAY RESULT IN PATCH LIFTING WHEN DEICER INFLATES.

CAUTION: MIXED REPAIR MATERIAL MUST BE USED WITHIN TWO (2) HOURS.

CAUTION: APPLICATION OF 74-451-AE KIT IS NOT RECOMMENDED BELOW 50°F (10°C), AS REPAIR MATERIAL MAY NOT DRY.

(a) Clean deicer surface to be repaired with detergent and hot water using lint-free cloth, to remove dirt, grease and cosmetic coatings. Repeat until oil and silicone contamination is removed.

(b) Isolate repair area with buffing template. Buff isolated area with Scotch Brite™ or equivalent.

(c) Wipe isolated area with lint-free cloth dampened with alcohol or equivalent cleaning solvent and allow to dry.

(d) Secure application template with hole centered over pinhole.

(e) Open foil overpack of 74-451-209 cement. Clip corner of Appli-pak pouch and squeeze cement into mixing cup. Make sure that all material from pouch is squeezed into mixing cup to assure proper mix ratio with accelerator.

CAUTION: RECAP ACCELERATOR TIGHTLY AFTER USE AS ACCELERATOR WILL HARDEN IF NOT SEALED COMPLETELY.

(f) Add seven (7) drops of P/N 74-451-201 accelerator. Mix thoroughly with wooden stir stick for one minute, making sure material on sides of cup is thoroughly mixed.

CAUTION: REMOVE APPLICATION TEMPLATE AS SOON AS REPAIR MATERIAL IS APPLIED AND LEVELED TO ASSURE SMOOTH APPEARANCE.

(g) Apply repair material into hole in application template with wooden stir stick. Push repair material firmly into pinhole. Level repair material with application tool.

NOTE: If repair material is spilled on undamaged deicer surface, remove immediately with Toluene.

(h) Allow two (2) hours dry time prior to flying the aircraft or inflating deicer.

(3) Loose Surface Ply in Dead Area (i.e. - non-inflatable area): Use P/N 74-451-AA patch kit to repair cuts, tears or ruptures that do not exceed 4” X 9” per Scuff (Surface) Damage, above. Surface ply peeling less than 4” X 9” can be repaired by trimming away loose surface material, and applying the 74-451-AA patch repair as follows:

(a) Peel and trim the loose surface ply to the point where the adhesion of surface ply to the deicer is good.

(b) Proceed with patch application per 74-451-AA patch kit application instructions, under Scuff Damage, above.

G. Resurfacing

P/N 74-451-L resurfacing kit provides an oil resistant urethane coating. The “L” Kit can be used to refurbish weathered deicers to extend service life, or to provide oil resistance to newly installed deicers. Apply the kit as follows:

CAUTION: AFTER KIT NO. 74-451-L IS APPLIED TO A NEOPRENE DEICER, THE DEICER SHOULD BE TREATED AS AN ESTANE® DEICER FOR MAINTENANCE PURPOSES.

CAUTION: DO NOT USE AGEMASTER® NO. 1 AFTER 74-451-L KIT APPLICATION.

NOTE: Deicers coated with 74-451-L kit may be repaired with 74-451-AA patch kit or 74-451-AE pinhole repair kit.

(1) 74-451-L kit covers 40-50 square feet of deicer surface.

(2) Wash deicer surface with mild soap and warm water to remove contamination.

(3) Remove ShineMaster® cosmetic treatment with ShineMaster® Prep cleaner.

(4) Lightly sand surface with Scotch Brite™ fine grade abrasive material or equivalent.

(5) Clean surface with clear water and dry with lint free cloths to remove soap residue and loose particles raised from sanding.

CAUTION: DO NOT USE MEK ON ESTANE® SURFACE DEICERS AS MEK WILL ATTACK ESTANE®.

(6) Wipe surface twice with cleaning solvent. Use MEK or Toluene on neoprene deicers; use only Toluene on Estane® deicers.

(7) Apply masking tape to leading edge against edges of deicer.

(8) Apply primer:

(a) For Estane® deicers, skip to step (9).

(b) For neoprene deicers only:

1) Mix primer coat with two (2) ounces of accelerator.

2) Apply one even coat of primer mixture to deicer surface. Brush perpendicular to deicer tubes using short strokes in one direction for smooth finish.

3) Let dry 30 minutes.

(9) Apply protective coat:

(a) For Estane® deicers, use protective coat alone.

(b) For neoprene deicers, mix protective coat with two (2) ounces accelerator.

(c) Apply one even coat of protective coat. If desired, a second coat of protective coat may be applied within fifteen (15) minutes to one (1) hour of first coat.

(10) Remove masking tape as application is made, about every three (3) feet.

(11) Aircraft may be flown and deicer inflated after 4 hours dry time.

(12) Do not use AgeMaster® No. 1 after 74-451-L kit application.

(13) Deicers coated with 74-451-L kit may be repaired with 74-451 AA patch kit or 74-451-AE inhole repair kit.

5. Pneumatic Deice System Adjustments and Test

A. S/N's 4636001–4636460, 4636463–4636651, less 4636481 and 4636633; 4692001 and up, less 4692141, 4692149 and 4692153.

(PIR-PPS60054-5, Rev. E., -6 Rev. E., -7. Rev. B.)

WARNING: DURING ANY ENGINE OPERATION SPECIFIED HEREIN, EXERCISE CAUTION TO AVOID HARM OR DAMAGE TO PERSONNEL AND EQUIPMENT BY THE PROPELLER AND PROPELLER BLAST.

NOTE: The vacuum is displayed on the MFD when either Avidyne Entegra or Garmin G1000 is installed. On G1000 equipped aircraft, Crew Alerting System (CAS) messages are displayed on the MFD.

(1) Start the engine and operate at 2,000 rpm.

(2) On G1000 equipped aircraft only: Remove the vacuum hoses (suction side) from both vacuum regulators. Verify the VACUUM 1 FAIL, VACUUM 2 FAIL Crew Alerting System (CAS) messages illuminate, the MASTER CAUTION indicator activates, and the MFD vacuum indicator text highlights in yellow. Reconnect vacuum hoses.

(3) Remove the vacuum hose (suction side) from the right vacuum regulator and adjust the left vacuum regulator to provide a vacuum gauge indication of:

(a) On conventional gauge or Avidyne equipped aircraft: 4.8 to 5.2 (S/N's 4636001 thru 4636020); 4.8 (min.) (S/N's 4636021, 4636132 thru 4636313); or, 4.5 to 5.2 (S/N's 4636022 thru 4636131, 4636314 and up, and S/N's 4692001 and up) inches Hg. of vacuum on the gauge and record the reading. In S/N's 4636021 and up, verify the VACUUM No. 1 INOP annunciator is illuminated and the VACUUM No. 2 INOP annunciator is not. In S/N's 4636001 thru 4636020, the left indicator button in the vacuum gauge shall be in (indicating pump is operating). Reconnect the right vacuum regulator.

(b) On G1000 equipped aircraft only: 4.5 to 5.2 inches Hg. of vacuum and record the reading. Verify the VACUUM 1 FAIL Crew Alerting System (CAS) message is illuminated and the VACUUM 2 FAIL message is not illuminated. Reconnect the right vacuum regulator.

(4) Remove the vacuum hose (suction side) from the left vacuum regulator and adjust the right vacuum regulator to the same value as the left regulator in (C) above.

(a) On conventional gauge or Avidyne equipped aircraft: In S/N's 4636001 thru 4636020, the right indicator button in the vacuum gauge shall be in (indicating pump is operating). In S/N's 4636021 and up, verify the VACUUM No. 2 INOP annunciator is illuminated and the VACUUM No. 1 INOP annunciator is not. Reconnect left vacuum regulator.

(b) On G1000 equipped aircraft only: Verify the VACUUM 2 FAIL Crew Alerting System (CAS) message is illuminated and the VACUUM 1 FAIL message is not illuminated. Reconnect the left vacuum regulator.

(5) In S/N's 4636021, 4636132 thru 4636313, with both regulators connected and both vacuum pumps operating, adjust the left and right regulators equally to obtain a vacuum gauge analog indication of 4.80 inches of Hg. Verify that the vacuum gauge indicates from 4.80 to 5.20 inches Hg. with engine operating at idle and 2,000 rpm.

In S/N's 4636001 and up (less 4636021 and 4636132 thru 4636313) and S/N's 4692001 and up, check that vacuum indication does not exceed 5.2 inches Hg. with both pumps operating and both regulators connected. If indication exceeds 5.2 inches, readjust the left and right vacuum regulators, per C and D above, to provide an indication of no more than 5.2 inches Hg.

(6) Check all surface boots and ensure they are down tight against the wing and tail surfaces.

(7) Using a vacuum (or combination pressure/vacuum) gauge connected to a hypodermic needle, measure the vacuum in the pneumatic system by inserting the needle in either right or left wing hose connections located on cabin side of the forward pressure bulkhead.

(8) Remove the vacuum gauge and install a pressure gauge at the same point. (This is not required if pressure/vacuum gauge is used.)

(9) Disconnect one of the vacuum pump pressure hoses from the pressure manifold (fwd of the firewall) and cycle the SURFACE DE-ICE switch ON to activate the system and check for air indication of 14 -0.5, +2 pounds per square inch on the pressure gauge. Reconnect the disconnected pressure hose and disconnect the other pump's pressure hose from the pressure manifold and repeat the system cycle to check for air indication of 14 -0.5, +2 psig. Reconnect pump pressure hose.

(10) Cycle the SURFACE DE-ICE switch ON to activate the system and check for air indication of 14 -0.5, +2 pounds per square inch on the pressure gauge.

(11) Visually check all boots for proper operation. All cells shall inflate fully. Each segment cycle time shall be 6 ± 2 seconds. Cycle sequence is tail, lower wing, upper wing.

(a) On conventional gauge or Avidyne equipped aircraft: The surface de-ice annunciator light shall illuminate and: in S/N's 4636021 and up and S/N's 4692001 and up, neither VACUUM No. 1 INOP nor VACUUM No. 2 INOP annunciator shall illuminate; in S/N's 4636001 thru 4636020, both left and right indicator buttons in the vacuum gauge shall be in during boot operation.

(b) On G1000 equipped aircraft only: The surf de-ice On message shall illuminate and both VACUUM 1 FAIL and VACUUM 2 FAIL messages shall not illuminate during boot operation.

(12) Remove the pressure or pressure/vacuum gauge connected in step (g) or (h) above, respectively.

(13) On G1000 equipped aircraft only:

(a) In the cabin, disconnect one of the three de-ice hoses at the fitting, and separate tubing such that air pressure exhausts into the cabin and not to the boots. (Fittings for the two wing hoses are located just aft of the right side circuit breaker waterfall. The fitting for the tail hose is located at the aft pressure bulkhead.)

(b) Cycle the SURFACE DE-ICE switch ON to activate the system. Verify that the SURF DE-ICE FAIL message illuminates. (The message willl take 20–30 seconds after the switch is cycled ON to illuminate).

(c) Reconnect the hose disconnected above.

(d) Cycle the SURFACE DE-ICE switch to ON to activate the system. Verify the SURF DE-ICE FAIL message extinguishes. (The message will take 20–30 seconds after the switch is cycled ON to extinguish).

(e) Repeat steps (a) through (d) for each of the two remaining de-ice hoses.

B. S/N's 4636633, 4636652 and up

(PIR-PPS60054-10, Rev. New.)

WARNING: DURING ANY ENGINE OPERATION SPECIFIED HEREIN, EXERCISE CAUTION TO AVOID HARM OR DAMAGE TO PERSONNEL AND EQUIPMENT BY THE PROPELLER AND PROPELLER BLAST.

(1) Remove the vacuum hoses (suction side) from both vacuum regulators.

(2) Start the engine and operate at 2000 revolutions per minute (rpm). Verify the VACUUM 1 FAIL and VACUUM 2 FAIL Crew Alerting System (CAS) messages illuminate, the MASTER WARNING/ CAUTION soft key illuminates, the MFD vacuum indicator text, pointer and number highlights in yellow, the vacuum reads 0, and a double chime is heard. Turn the engine off and reconnect left vacuum hose.

(3) Start the engine and operate at 2000 rpm. Check that the left vacuum regulator provides a vacuum gauge indication of 4.5 to 5.2 inches of mercury (inHg) vacuum. If not, turn the engine off, adjust the regulator, and try again. Record the value on the traveler. Verify the VACUUM 1 FAIL CAS message is illuminated and the VACUUM 2 FAIL CAS message is not illuminated. Turn the engine off. Reconnect right vacuum hose and remove the vacuum hose (suction side) from the left vacuum regulator.

(4) Start the engine and operate at 2000 rpm. Check that the left vacuum regulator provides a vacuum gauge indication the same as in step d. If not, turn the engine off, adjust the regulator, and try again. Record the value on the traveler. Verify the VACUUM 2 FAIL CAS message is illuminated and the VACUUM 1 FAIL CAS message is not illuminated. Turn the engine off and reconnect the left vacuum hose.

(5) Start the engine and operate at 2000 rpm. Verify that vacuum indication does not exceed 5.2 inHg with both pumps operating and both regulators connected. If indication exceeds 5.2 inHg, stop the engine and readjust the left and right vacuum regulators per steps (3) and (4) above to provide an indication not to exceed 5.2 inHg.

(6) Check all surface boots and ensure they are down tight against the wing and tail surfaces. Turn the engine off.

(7) Install a pressure or pressure/vacuum gauge tee'd into the line at the bulkhead fitting at the lower right portion of firewall (RBL 15.50, FS 79.00, WL 52.30) and disconnect the left vacuum pump pressure hose from the pressure manifold.

(8) Start the engine and operate at 2000 rpm. Turn the SURFACE DE-ICE switch on to activate the system. Verify indication of 14.0 +2.0 – 0.5 psig on the pressure gauge. Record the value on the traveler. Turn SURFACE DE-ICE switch off. Turn the engine off and reinstall the left vacuum pump pressure hose and remove the right vacuum pump pressure hose from the pressure manifold.

(9) Start the engine and operate at 2000 rpm. Turn the SURFACE DE-ICE switch on. Verify indication of 14.0 +2.0 -0.5 psig on the pressure gauge. Record the value on the traveler. Turn off SURFACE DE-ICE switch and turn the engine off and Reconnect the right vacuum pump pressure hose.

(10) Start the engine and operate at 2000 rpm. Turn the SURFACE DE-ICE switch on. Verify indication of 14.0 +2.0 -0.5 psig on the pressure gauge. Record the value on the traveler.

(11) Visually check all boots for proper operation. All cells shall inflate fully. Each segment cycle time shall be 6 ± 2 seconds. Cycle sequence is; tail, lower wing, upper wing, and then a 36 second dwell period. Verify SURF DEICE ON white CAS message is illuminated and both VACUUM 1 FAIL and VACUUM 2 FAIL white CAS messages are not illuminated. Turn the engine off and remove the pressure or pressure/vacuum gauge connected in step (7).

(12) In the cabin, disconnect the upper wing de-ice hose at the fitting in the cabin on the right side aft of the waterfall panel and separate tubing such that air pressure exhausts into the cabin and not to the boots.

(13) Start the engine and operate at 2000 rpm. Turn the SURFACE DEICE switch on. Verify that the SURF DEICE FAIL amber CAS message is illuminated. Turn the SURFACE DEICE switch off during the dwell period and Reconnect the hose disconnected in step (12).

(14) Turn the SURFACE DEICE switch on. Verify the SURF DEICE FAIL amber CAS message does not illuminate. Turn the SURFACE DEICE switch off during the dwell period.

(15) In the cabin, disconnect the lower wing de-ice hose at the fitting in the cabin on the right side aft of the waterfall panel and separate tubing such that air pressure exhausts into the cabin and not to the boots.

(16) Turn the SURFACE DEICE switch on. Verify that the SURF DEICE FAIL amber CAS message is illuminated. Turn the SURFACE DEICE switch off during the dwell period and reconnect the hose disconnected in step (15).

(17) Turn the SURFACE DEICE switch to on. Verify the SURF DEICE FAIL amber CAS message does not illuminate. Turn the SURFACE DEICE switch off during the dwell period.

(18) In the cabin, disconnect the tail de-ice hose at the fitting in the aft cabin and separate tubing such that air pressure exhausts into the cabin and not to the boots.

(19) Turn the SURFACE DEICE switch on. Verify that the SURF DEICE FAIL amber CAS message is illuminated. Turn the SURFACE DEICE switch off during the dwell period.

(20) Reconnect the hose disconnected in step (18).

(21) Turn the SURFACE DEICE switch to on. Verify the SURF DEICE FAIL amber CAS message does not illuminate. Turn the SURFACE DEICE switch off during the dwell period. Turn the engine off.

6. SMARTboot™ Ice Detection System (S/N's 4636132 thru 4636313 only)

A. Description

The optional pneumatic wing deice system, as installed in S/N's 4636132 thru 4636313 only, incorporates the B.F. Goodrich SMARTboot™ Ice Detection System. This system advises the pilot of icing conditions on the empennage of the airplane. Electrical sensors integrated directly into the surface of the pneumatic de-icers on the vertical fin and right horizontal stabilizer advise the presence of ice and appropriate time to de-ice.

NOTE: The system is installed as and performs only an advisory function. The addition of the B.F. Goodrich SMARTboot™ Ice Detection System does not change the form, fit or function of the other approved de-icing systems installed in the Malibu Mirage.

NOTE: If desired the B.F. Goodrich SMARTboot™ Ice Detection System may be removed by installing Piper Kit No. 88549-001.

36-inch electrical sensors are vulcanized directly into, and flush with, the surface ply of the pneumatic de-icer boots for the vertical fin and the right horizontal stabilizer. These sensors are located in the leading edge of the de-icer boots, are made from the same type of elastomeric material, and do not change the ice adhesion or the erosion resistance of the boots. Additionally, the sensors are flush with the surface of the de-icer boots and there is no change to airflow or icing characteristics of the leading edge surfaces.

All wiring from the sensor to the electrical connector is contained within the de-icer and routed through non-inflatable regions of the boot to the same location as the air connection for the de-icer to eliminate the need for any additional installation holes. Inside the wing, the wiring harness is again routed through the same space as the air lines.

The presence of ice and appropriate time to deice are measured by the sensor installed within the vertical fin deicer only. The sensor within the right horizontal stabilizer deicer advises the status of the detection system (ice detect fail annunciator).

These annunciators located within the annunciator panel make up the visual cues of the deice detector / deicer system. The "select de-ice" annunciator illuminates when approximately 1/4 inch of ice has accumulated on the sensor located on the vertical fin.

The surface de-ice annunciator will illuminate to advise the pilot of proper boot inflation. The ice detect fail annunciator will illuminate when a fault is detected in the ice detection system.

To conduct a self test of the system while on the ground turn the battery master switch on. This will supply power to the system for a complete system self test. During the self test, the select de-ice and the ice detect fail annunciators will flash momentarily in sequence, activate all at once, and then extinguish.

The green "select De-ice" annunciator is activated when the ice thickness reaches approximately 1/4 inch or greater over the surface of the sensor in the vertical fin boot. When this occurs, the pilot may choose to inflate the deice boots by pressing the "surface de-ice" switch or choose to wait until more ice forms on the wings.

If the ice is removed from the empennage following inflation, the "select De-ice" annunciator will extinguish. If less than 90% of the ice cap located over the sensor within the right horizontal deicer has been removed, the "select De-ice" annunciator will remain illuminated. If a fault should develop with either the sensor or controller, the amber "ice detect fail" annunciator will illuminate.

Because the sensors are installed in the empennage and the smaller radius surfaces there tend to accumulate ice faster, the "select De-ice" annunciator will occasionally illuminate before the main wing has accumulated 1/4 to 1/2 inch thickness of ice.

B. Verification

NOTE: This test can be performed without engine operation.

(1) Pull ICE DETECTOR circuit breaker.

(2) Disconnect the OAT sensor located in the vertical fin fairing. Install a 90 ohm (± 5%) resistor across pins A and B of the connector of the 3E3176-1 wire harness. Install a wire between pins B and C so that they are common.

(3) Activate the system circuit breaker and check the operation through the cockpit annunciator panel.

NOTE: Following activation of the system circuit breaker, the amber SELECT DE-ICE and amber ICE DETECT FAIL annunciators will flash momentarily in sequence, activate all at once, and then extinguish as the controller performs its self-check.

(4) Push the annunciator test switch and verify that the amber SELECT DE-ICE, amber ICE DETECT FAIL, and the green SURFACE DE-ICE lights illuminate.

(5) Align the five wires of the 3E3302-1 test strip over the five sensors in the vertical fin boot and tape in place or thoroughly saturate two cloths (small dish towel size) with tap water and place over the vertical fin sensor.

(6) The amber SELECT DE-ICE light should illuminate.

NOTE: The amber SELECT DE-ICE light may take approximately ten minutes to illuminate while the controller steps through the ice thickness measurement functions. Make sure the wet cloths remain in contact with the sensor and remain saturated or the 3E3302-1 test strip wires remain aligned with the fin boot sensor wires.

(7) Disconnect connector J1 from the controller box. Be careful not to disconnect connector J2 from the controller box. The amber ICE DETECT FAIL light should begin flashing immediately.

(8) Reconnect the J1 connector. Remove the resistor and shunt wire from the 3E3181-1 connector, reattach the OAT sensor. Deactivate and reset the system circuit breaker. The amber ICE DETECT FAIL light should extinguish.

(9) Remove the wet cloths or 3E3302-1 test strip from the vertical fin.

(10) Return the airplane's wiring, de-ice plumbing, and panels to original configuration.

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PITOT AND STATIC

1. Heated Pitot Head

A. Description

The standard pitot ice protection consists of a heated pitot head located on the bottom of the wing: on the left side (in both PA-46-350P and PA-46R-350T airplanes) and right side (in all PA-46-350P airplanes and PA-46R-350T airplanes euipped with Garmin G1000). The heated pitot head(s) are activated by a switch on the De-ice Switch Panel (S/N's 4636001 thru 4636131 only) or Environmental / Deice Switch Panel (S/N's 4636132 and up and S/N's 4692001 and up), a 10-amp circuit breaker for each heated pitot head, a relay and the circuit wiring.

B. Removal, Installation and Testing

See 34-10-00.

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WINDOWS AND WINDSHIELDS

Heated Windshield

A. Description

See “Figure 1”.

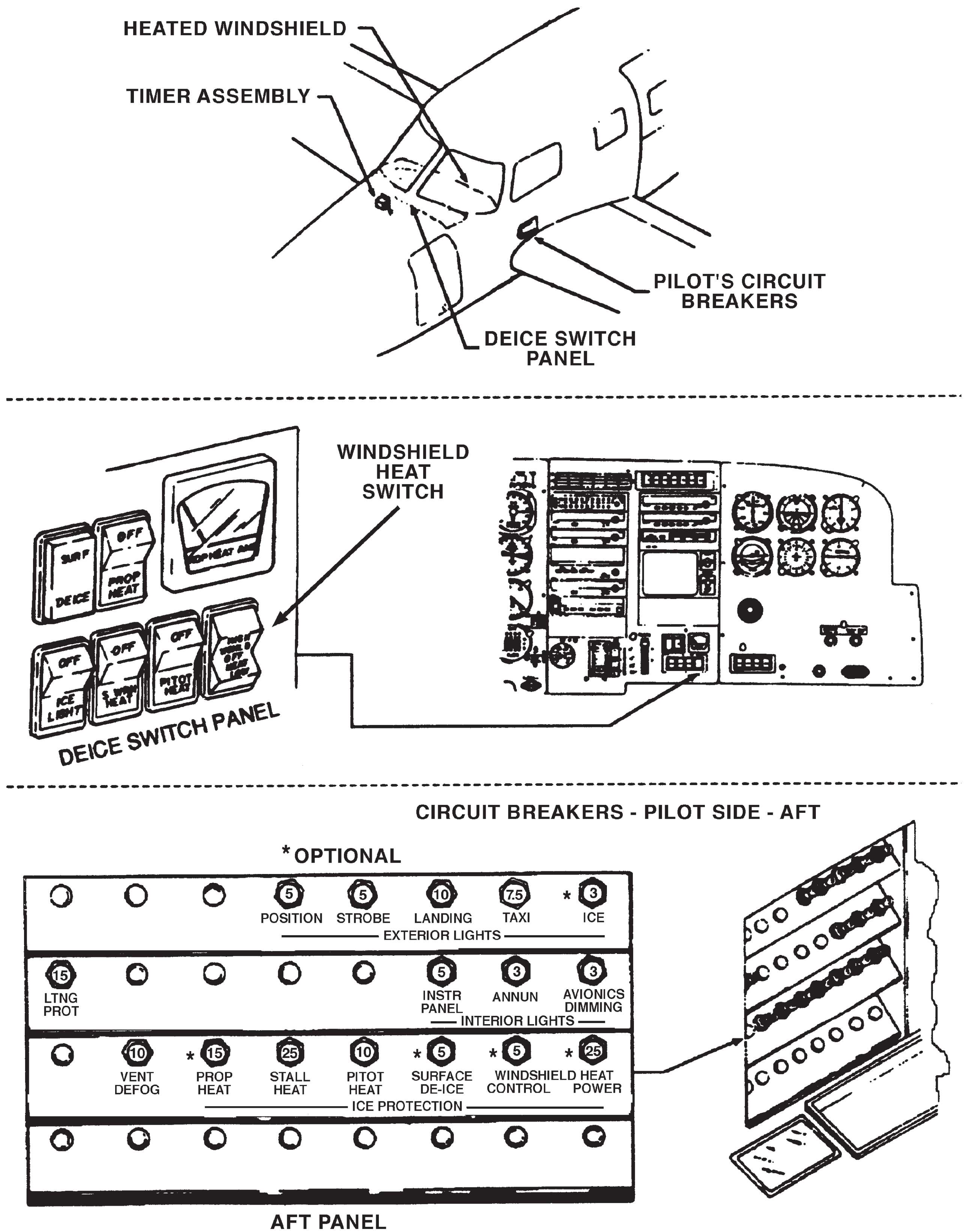
When installed, the heated windshield consists of an electrically heated left windshield. Control is provided by a switch on the De-ice Switch Panel (S/N’s 4636001 thru 4636131 only) or Environmental / Deice Switch Panel (S/N’s 4636132 and up and S/N’s 4692001 and up). Circuit protection is provided by the windshield heat CONTROL and POWER breakers in the ICE PROTECTION section of the circuit breaker panel.

B. Troubleshooting

See “Chart 1”.

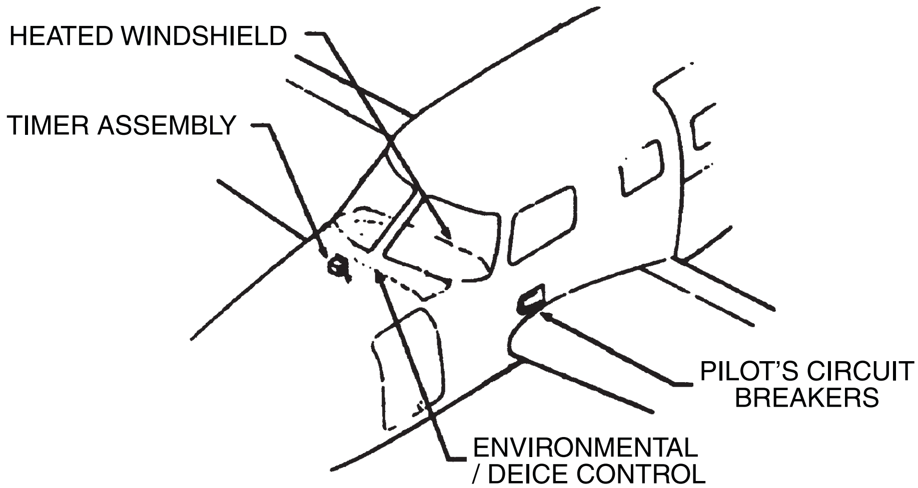
CHART 1
TROUBLESHOOTING HEATED WINDSHIELD SYSTEM

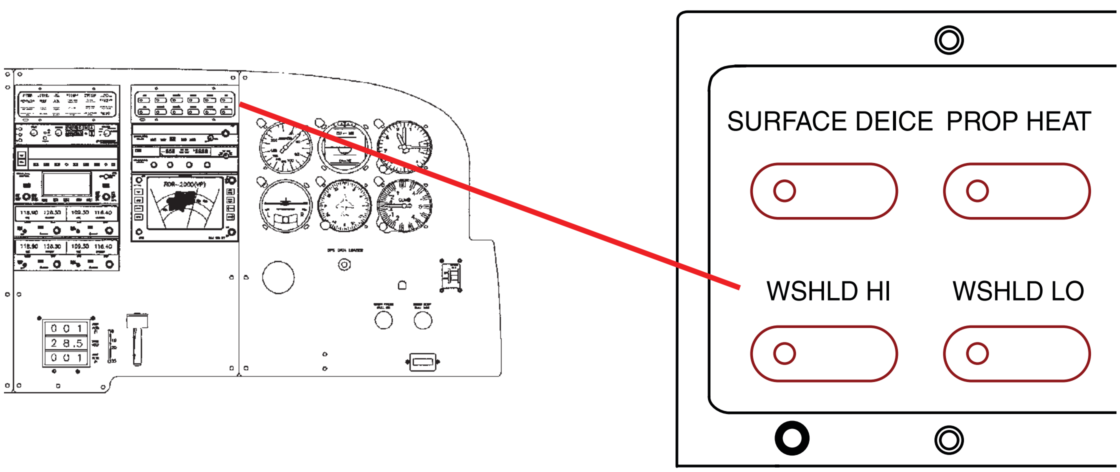
Trouble	Cause	Remedy
Windshield fails to heat or no change between LOW and HIGH position	Circuit breaker tripped	Reset circuit breaker.
	Relay failure	Replace relay.
	Controller failure	Replace controller.
	Broken wire in harness.	Repair broken wire.
	Cannon plug loose or dirty.	Clean and re-engage cannon plug.
	Switch defective.	Replace switch.
	Poor connection on terminal strip.	Tighten screws on terminal strip in console.
CAUTION: SEVERE OVERHEATING OF WINDSHIELD COULD RESULT IN WARPING AND DISTORTION.		
Windshield overheats	Timer shorted.	Replace timer-temperature control.
	Windshield heat left on too long.	Manually turn windshield heat off.



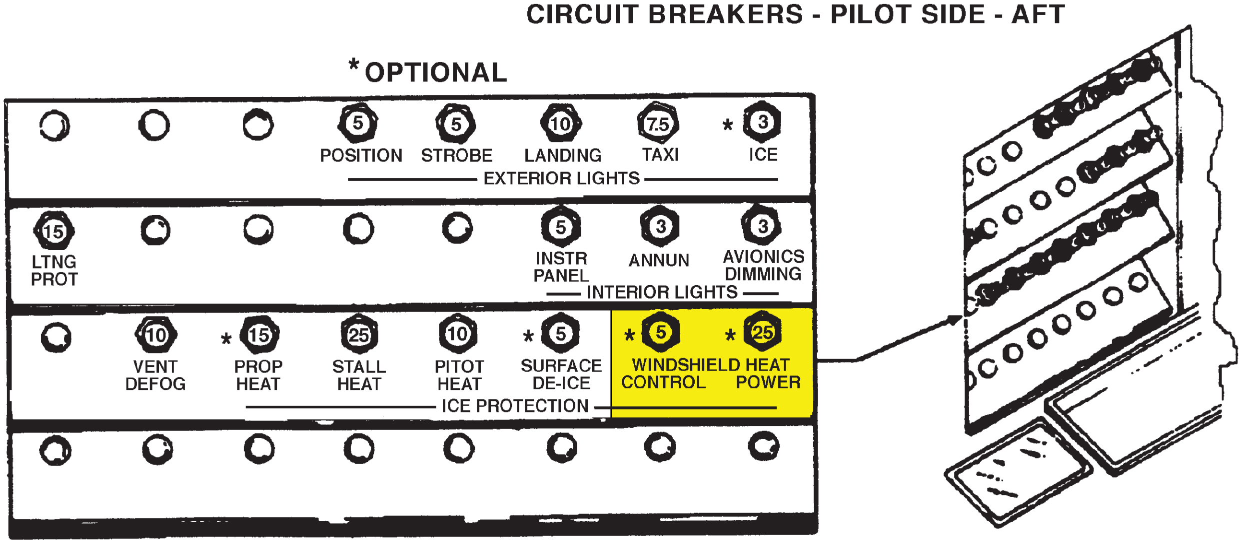
Effectivity
4636021 thru 46360131 shown

Windshield Heat
Figure 1 (Sheet 1 of 5)





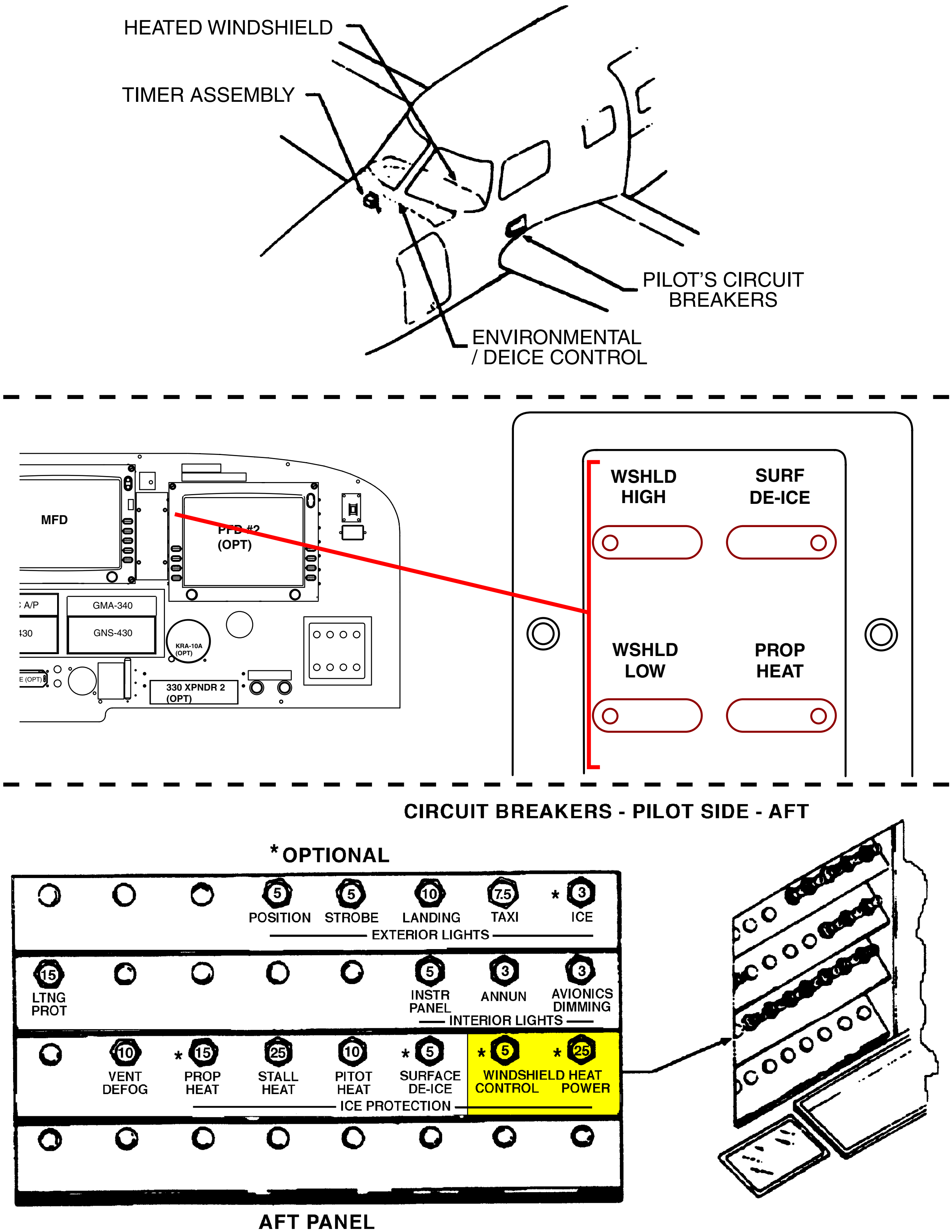
CIRCUIT BREAKERS - PILOT SIDE - AFT



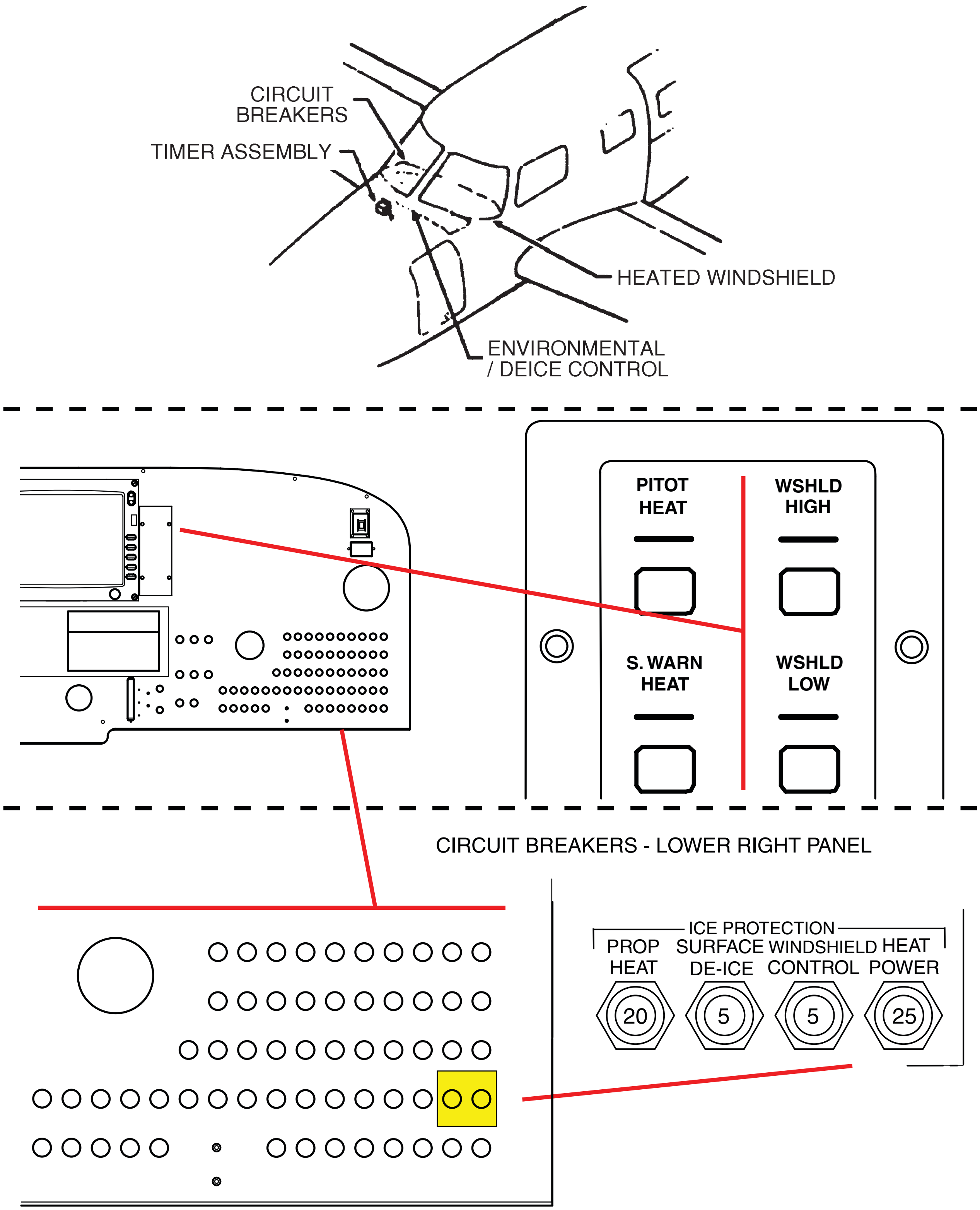
AFT PANEL

Windshield Heat
Figure 1 (Sheet 2 of 5)

Effectivity
4636132 thru 4636374

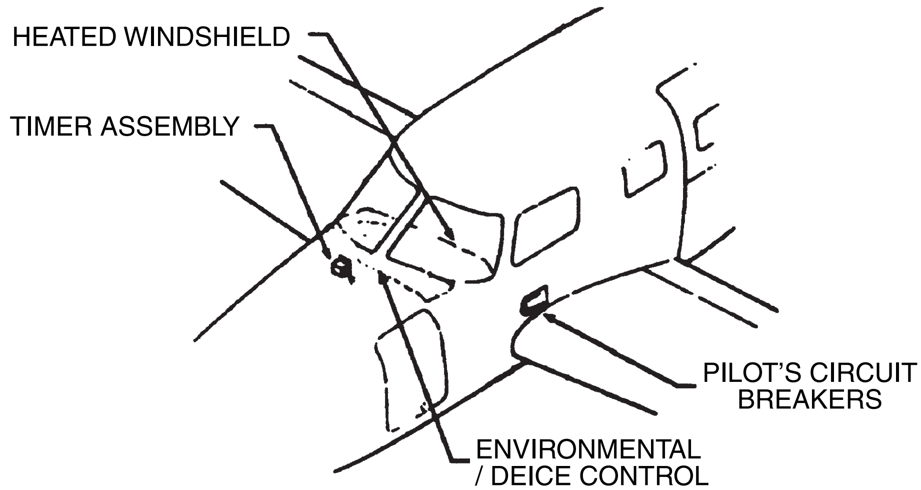


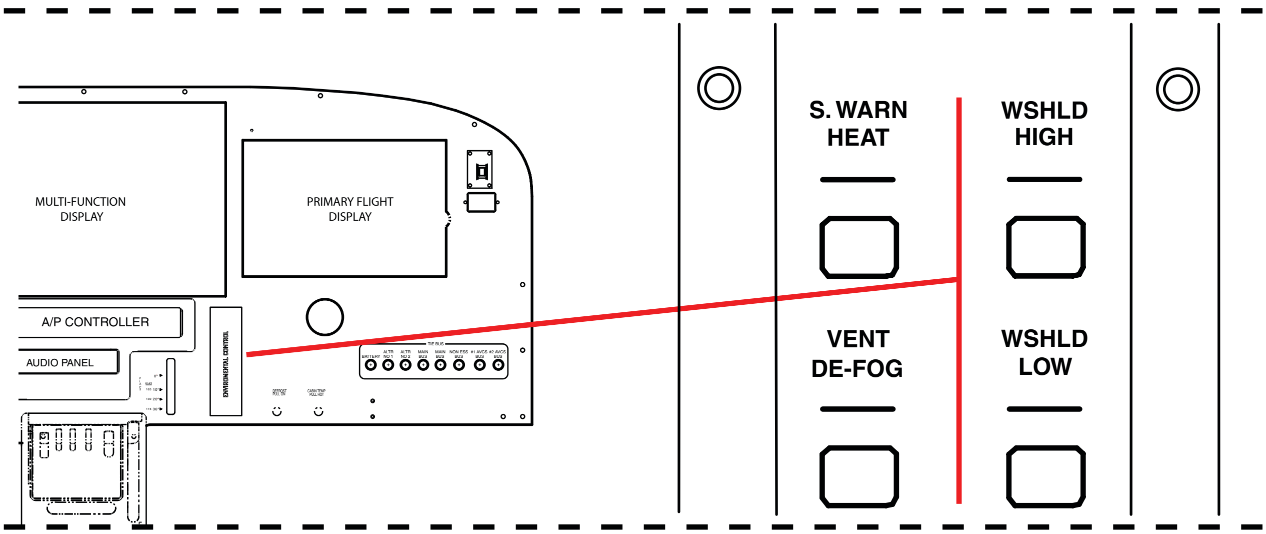
Windshield Heat
Figure 1 (Sheet 3 of 5)



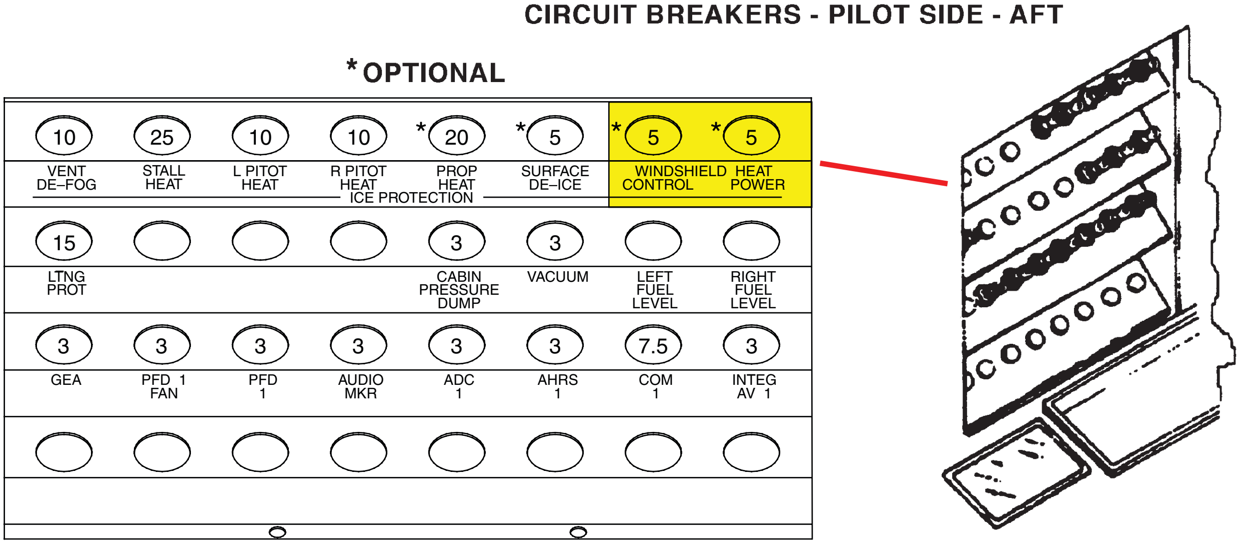
Windshield Heat
Figure 1 (Sheet 4 of 5)

Effectivity
4692001 and up





CIRCUIT BREAKERS - PILOT SIDE - AFT



AFT PANEL

Windshield Heat
Figure 1 (Sheet 5 of 5)

Effectivity
with G1000 Installation

30406

C. Testing

CAUTION: DO NOT USE IN-COCKPIT AMMETER READINGS TO ASSESS THE SERVICEABILITY OF HEATED WINDSHIELDS INSTALLED IN THESE AIRPLANES.

Do not use In-cockpit ammeter readings to assess the serviceability of heated windshields installed in these airplanes. The only Piper approved method to determine the serviceability of the heated windshield is as follows:

NOTE: This test is to be performed in ambient temperatures of 75° F +/– 10° (24° C +/– 5.56°).

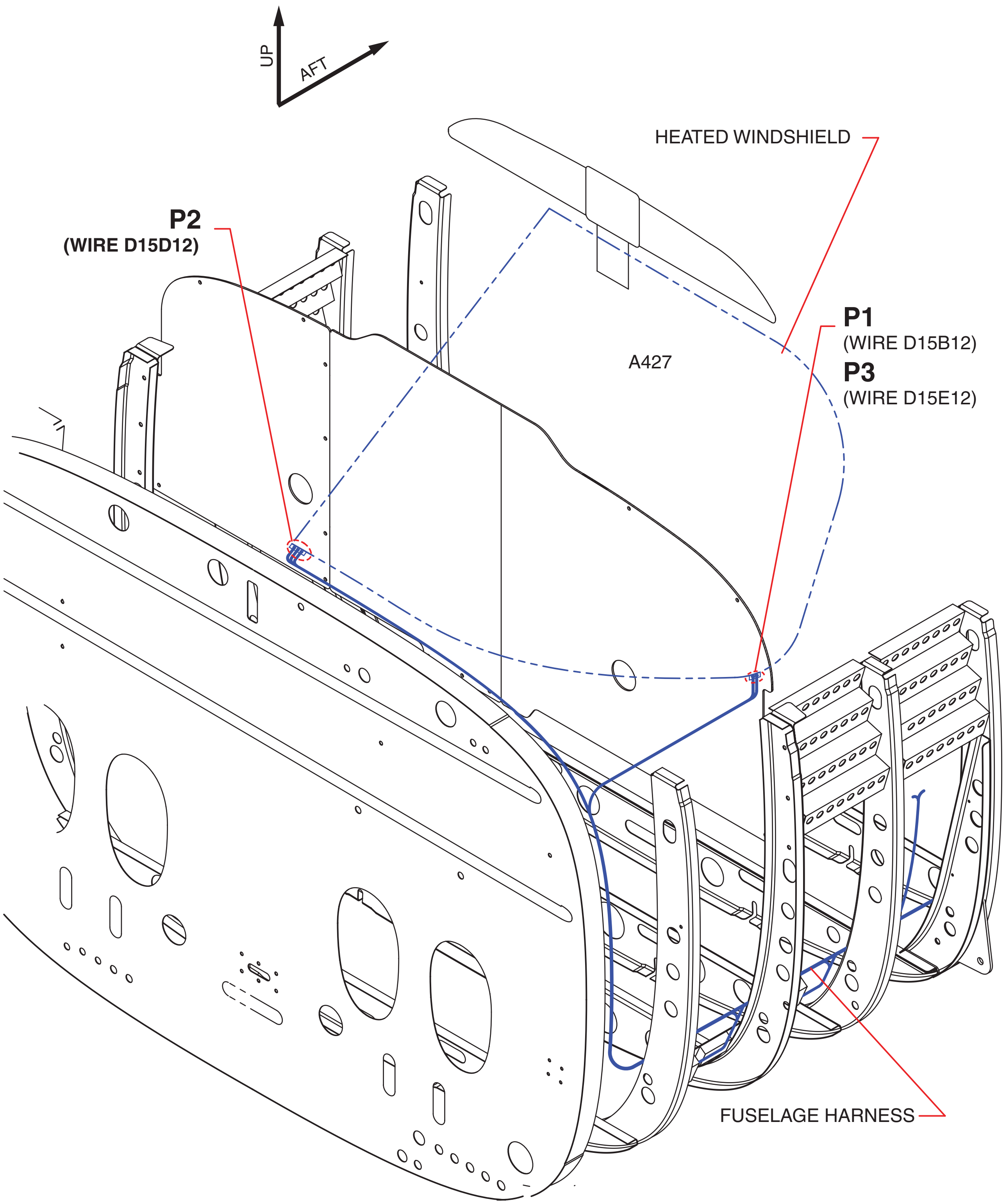
(1) Remove the glareshield to access the heated windshield terminal blocks. See “Figure 2” on page 30408 for terminal block locations.

(2) Resistance is to be tested with power removed from the heated windshield. When the windshield terminal blocks are exposed:

  (a) Measure resistance between terminal P1 and P2. Resistance must be 1.15 ohms +/– 10%.

  (b) Measure resistance between terminal P1 and P3. Resistance must be 1.58 ohms +/– 10%.

(3) If the measurements fail to meet specified values in either step (a) or (b) above, then the windshield must be replaced.



GLARESHIELD REMOVED TO ACCESS HEATER TERMINALS

Heated Windshield Terminal Block Locations
Figure 2

30408

PROPELLERS

1. Propeller Heat System

A. Description and Operation

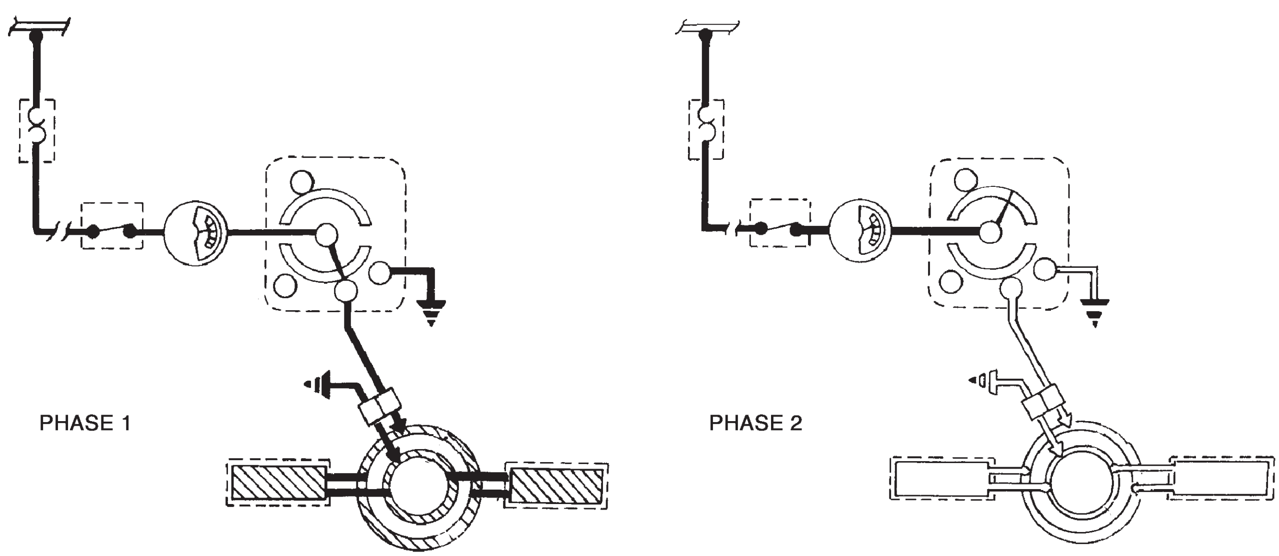
The propeller heat system installed on this airplane uses single element deice boots. In this system each deicer has one electrothermal heating element. When the PROP HEAT switch is turned ON, the timer directs power through the modular brush assembly and slip ring to all the heating elements on the propeller for approximately 90 seconds. The timer then switches power off for approximately 90 seconds (see “Figure 1” and 91-30-60). This cycle will continue as long as the PROP HEAT switch is in the ON position except in S/N’s 4636132 and up, S/N’s 4692001 and up.

In S/N’s 4636132 and up, S/N’s 4692001 and up, the PROP HEAT system employs a Moritz timer and a squat switch relay to create a ground test mode which interrupts power to the boots after an initial 30-45 seconds as long as the squat switch indicates weight on the wheels. During the 30-45 second test cycle, the PROP HEAT switch will flash at a fast rate and the ship’s ammeter will show an increase. In flight, the PROP HEAT switch flashes at a slow rate during the 90 second off cycle and is continuously illuminated during the 90 second ON cycle.

The propeller heat system consists of the following:

In S/N’s 4636001 thru 4636131: electrically heated deicers bonded to the propeller blades; slip ring assemblies connected to the propeller hub to distribute power to the propeller deicers; a modular brush assembly which transfers electrical power to the rotating slip rings; a timer to cycle power to the deicers; a propeller heat ammeter to indicate that the system is functioning correctly; an external ammeter shunt, a manual ON-OFF switch; and a circuit breaker.

In S/N’s 4636132 and up, S/N’s 4692001 and up: electrically heated deicers bonded to the propeller blades; slip ring assemblies connected to the propeller hub to distribute power to the propeller deicers; a modular brush assembly which transfers electrical power to the rotating slip rings; when equipped with the optional composite 3-blade propeller only - a metal oxide varistor (MOV) module to protect the propeller; a timer to cycle power to the deicers; an LED equipped manual ON-OFF switch; a circuit breaker; and a squat switch relay.



Prop Heat Cycle Sequence
Figure 1

CHART 1 (Sheet 1 of 4)
TROUBLESHOOTING PROPELLER HEAT SYSTEM

Trouble	Cause	Remedy
Ammeter shows zero current both phases of the timer cycle.	Tripped circuit breaker switch.	Locate and correct short before setting circuit breaker.
	No power from airplane.	If no voltage into switch, locate and correct open.
	Circuit breaker or switch faulty.	If no voltage at C/ B output with voltage at input and C/B does not reset, replace C/B. If voltage is OK at output, check switch in same manner. If voltage is OK at switch output, go to next step.
	Ammeter faulty. (If either or both deicers heat with ammeter at zero, replace the ammeter.)	Test for voltage up to and out of ammeter. If low output and input satis-factory, replace ammeter. If no voltage to ammeter, locate and fix open between switch and ammeter.
	Open ammeter to timer.	Disconnect harness at timer and check voltage at Pin B (of harness) to ground. If none, locate and correct open.
Ammeter shows normal current part of cycle, zero current rest of cycle.	Open in wiring between timer and brush block assembly.	Use heat test to find deicers not heating and test for voltage on that contact of wire harness plug. (At brush block assembly.) If zero after one minute, locate and fix open in wiring from timer to wire harness plug.
	Open between brush block assembly and deicer lead straps.	If there is voltage to brush block wire harness plug, try voltage at junction to deicer lead and slip ring lead. If no voltage, find and correct open in wiring within brush block or in contact of brush to slip ring.

CHART 1 (Sheet 2 of 4)
TROUBLESHOOTING PROPELLER HEAT SYSTEM

Trouble	Cause	Remedy
Ammeter shows normal current part of cycle, low current rest of cycle.	Open or high resistance in deicer or slip ring leads.	Disconnect deicer leads to check resistance. If not OK, replace faulty deicer. If satisfactory, locate and fix open in slip ring leads. Check deicer wire harness leads for continuity. Ohmmeter needle must not flicker when leads are stretched or flexed. Replace as required.
	High resistance in circuit with low current.	Check for contact of brush to slip ring. Check wiring from timer to deicers for loose or corroded connections and partially broken wiring. Correct as required.
Ammeter shows low current over entire cycle.	Aircraft voltage low.	Check voltage into switch.
	Ammeter faulty.	Test for voltage up to and out of ammeter. If low output and input satisfactory, replace ammeter. If no voltage to ammeter, locate and fix open between switch and ammeter.
	High resistance up to timer.	Check for partially broken wire, loose or corroded connection in wiring from aircraft supply to timer input.
Ammeter shows excess current over entire cycle.	Ammeter faulty.	Test for voltage up to and out of ammeter. If low output / input satisfactory, replace ammeter. If no voltage to ammeter, locate and fix open between switch and ammeter.
	Ground between ammeter and timer.	Disconnect harness at timer. Use ohmmeter to check Pin B (of harness) to ground. If ground indicated, locate and correct.

CHART 1 (Sheet 3 of 4)
TROUBLESHOOTING PROPELLER HEAT SYSTEM

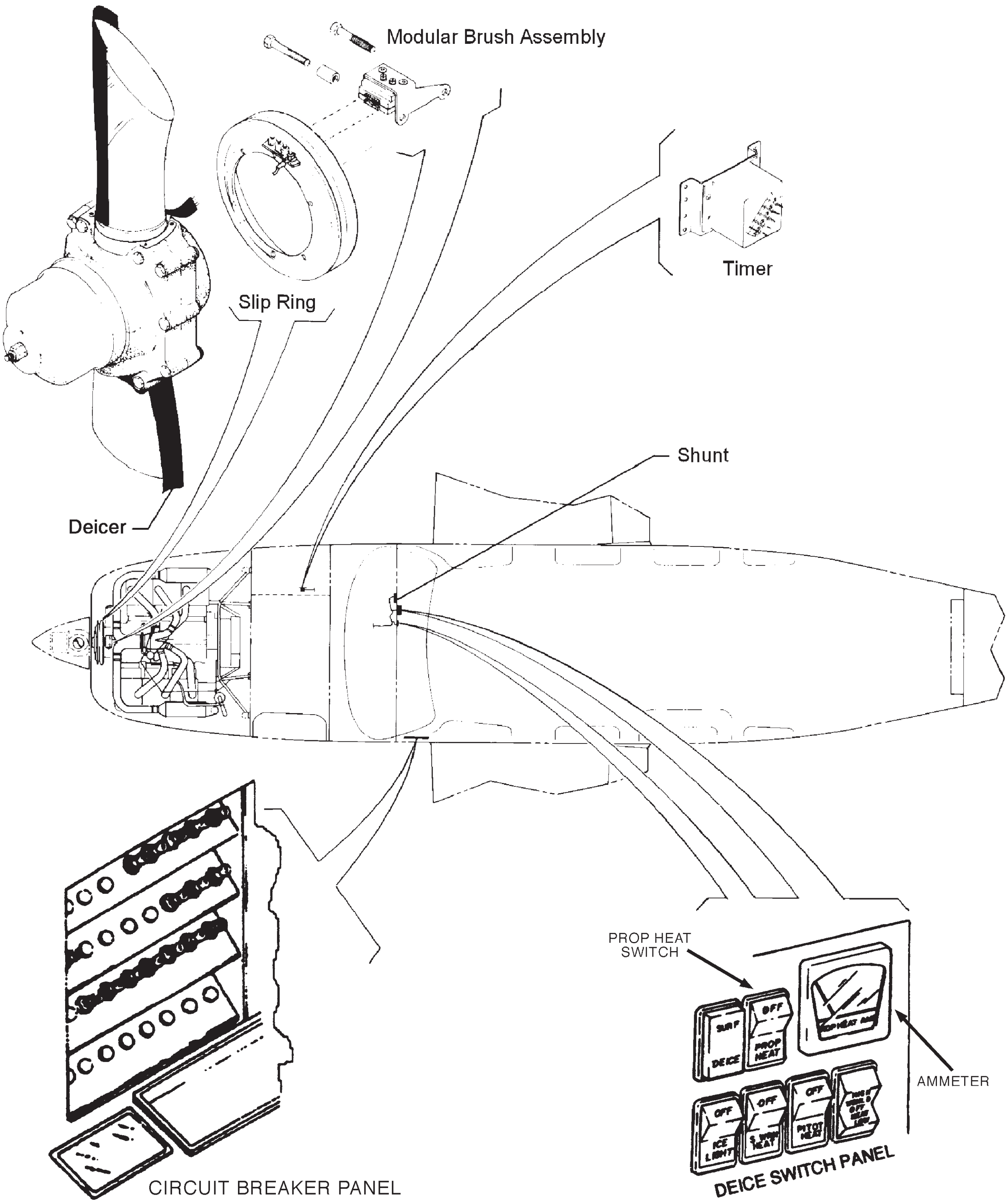
Trouble	Cause	Remedy
Ammeter shows normal current part of cycle, excess current rest of cycle.	Ground between timer and brush block.	Disconnect leads at brush block and with ohmmeter check from power leads to ground. If ground is indicated, locate and correct.
	Ground between brush block and deicers.	If no short exists at brush - slip ring contact, check for ground from slip ring lead to propeller assembly while flexing slip ring and deicer leads. If a ground is indicated, locate and correct.
	Short between two adjacent circuits.	Check for cuts or low resistance between circuits. If any, locate and correct.
	Timer faulty.	Test timer.
Ammeter does not “flick” approximately every 90 seconds.	Timer ground open, timer not cycling.	Disconnect harness at timer and check with ohmmeter from Pin G (of harness) to ground. If no circuit, fix open per schematic diagram.
	Timer contacts are welded (caused by short circuit in system).	Test timer. If timer does not cycle with voltage at Pin B, replace timer but be sure short causing original failure has been located and corrected.
Ammeter flicks between 90 second phase periods.	Loose connection between aircraft power supply and timer input.	Trace wiring from power source to timer input to ensure that the electrical contacts at each connection in the circuit are good.
	Loose or poor connection timer to deicers.	If trouble occurs in part of cycle, find which deicer is affected and check for rough or dirty slip rings causing brush to “skip.” Also, check for loose or poor connection. (If both deicers are affected, check ground circuit.)
	Timer cycles erratically.	Test timer.

CHART 1 (Sheet 4 of 4)
TROUBLESHOOTING PROPELLER HEAT SYSTEM

Trouble	Cause	Remedy
Radio noise or interference with deicers on.	Brushes “arcing.”	Check brush alignment as shown in Figures 3 and 5. Look for rough or dirty slip rings. If either condition exists, clean, machine or replace slip ring assembly, as required. Check slip ring alignment.
	Loose connection.	Refer to “Ammeter flicks between 90 second phase period.”
	Switch or circuit breaker faulty.	Place jumper wire across switch or circuit breaker. If radio noise disappears, replace the switch or circuit breaker.
	Wiring located too close to radio equipment or associated wiring.	Relocate deicer wiring at least 8 inches away from radio equipment and wiring.
Rapid brush wear or frequent breakage.	Brush-block out of alignment.	Check brush alignment. Correct as required.
	Slip ring wobbles.	Check slip ring alignment with dial indicator as shown in Figure 5.
NOTE: Ammeter is displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.		

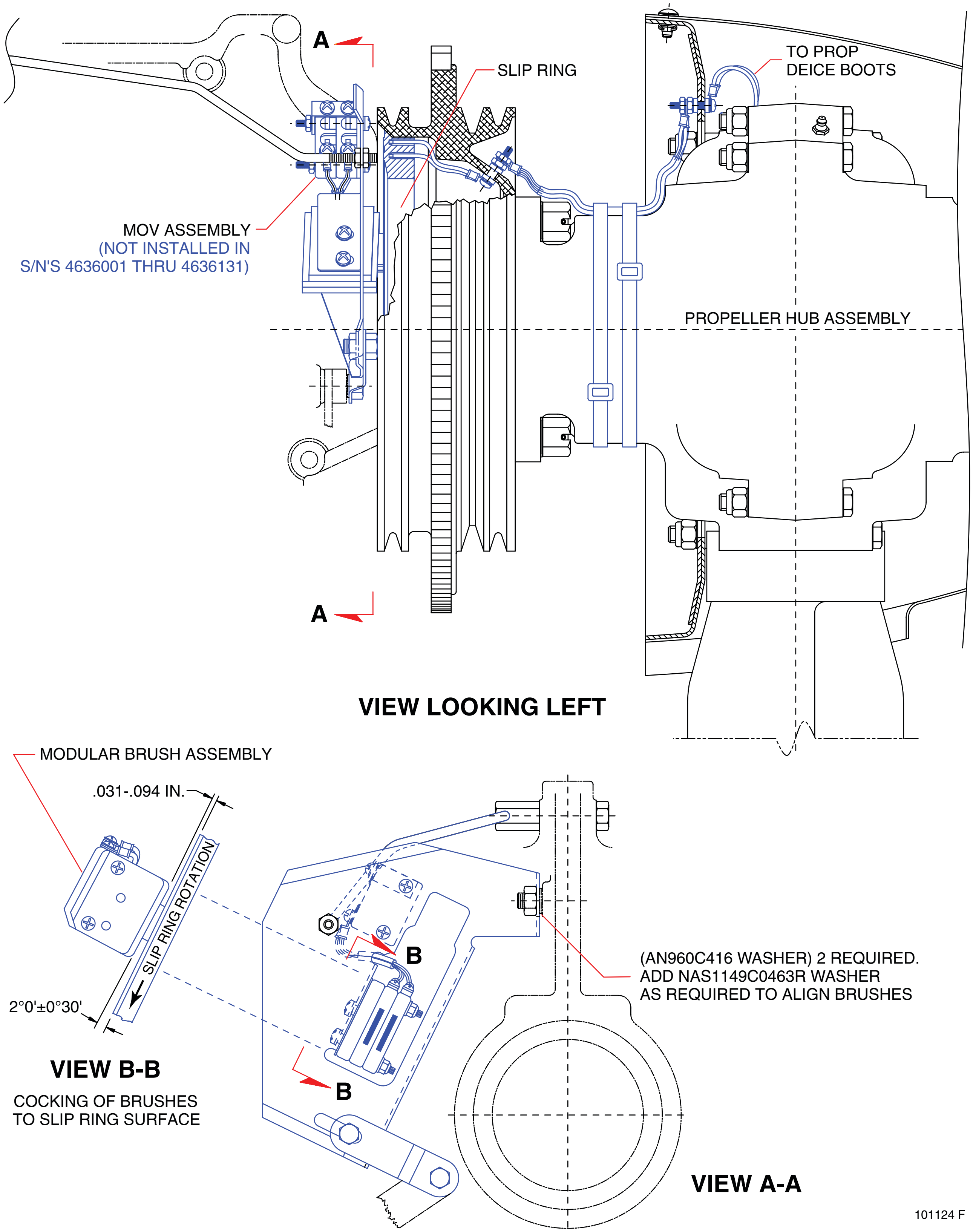
B. Troubleshooting

See “Chart 1”. See also Testing, below.



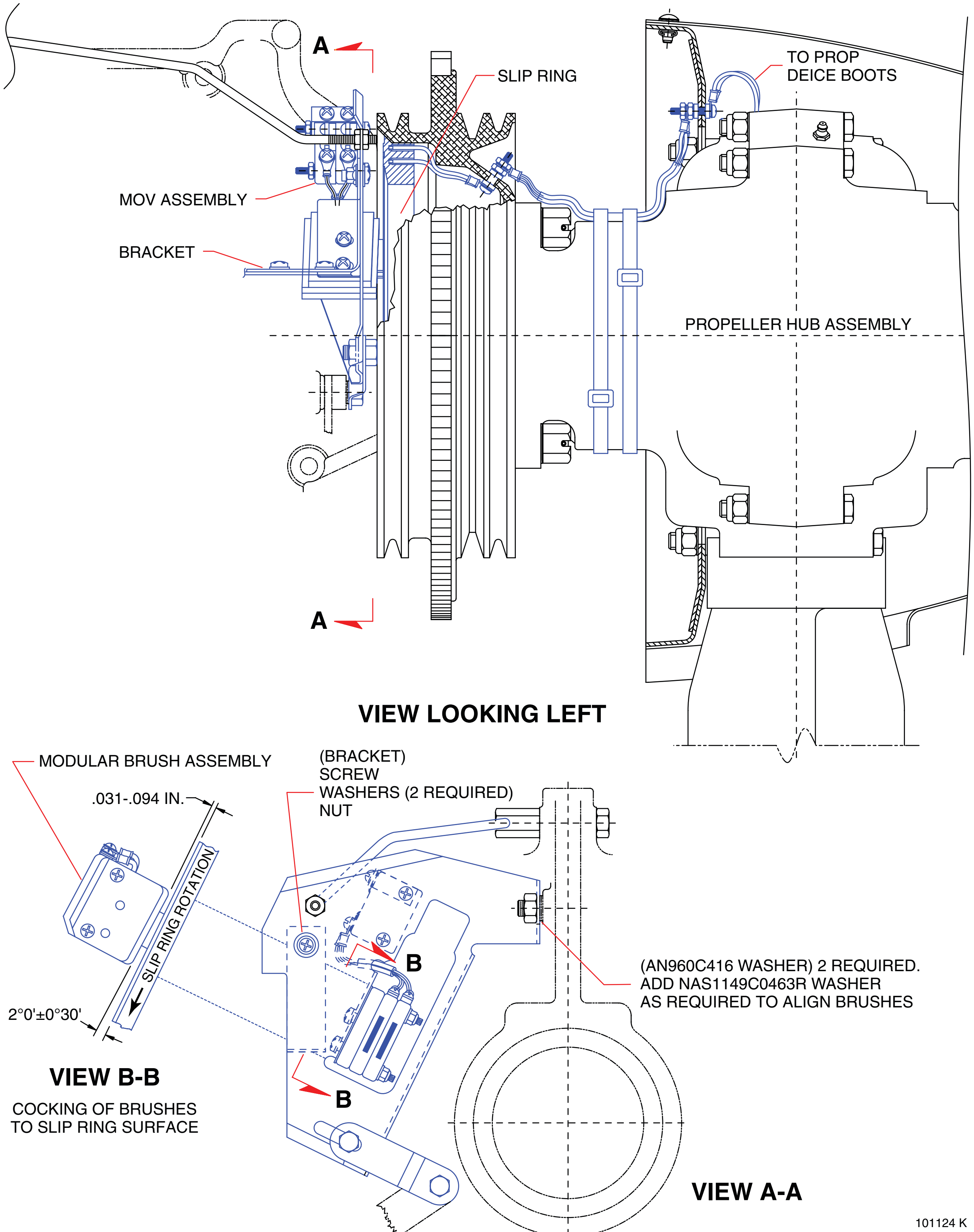
Effectivity
4636001 thru 4636131 shown

Propeller Heat System Installation
Figure 2



Effectivity
4636001 thru 4636528
4692001 thru 4692177

Modular Brush Assembly Alignment
Figure 3 (Sheet 1 of 2)



Effectivity
4636529 and up
4692178 and up

Modular Brush Assembly Alignment
Figure 3 (Sheet 2 of 2)

2. Components

NOTE: Do not attempt internal repairs of the timer, ammeter or switch. If inoperative, these components must be replaced. For any other repair or maintenance problems not covered herein, inquire at De-Icing & Specialty Systems Division of the Goodrich Corporation or Hartzell Propeller, depending on which components are installed on your airplane.

A. Deicers (i.e. - Heater Boots)

WARNING: CEMENTS AND SOLVENTS USED TO REMOVE AND INSTALL DEICERS ARE EXTREMELY FLAMMABLE AND TOXIC. EXTINGUISH OPEN FLAMES. AVOID SPARKS. USE IN WELL-VENTILATED AREA. AVOID SKIN CONTACT AND/OR PROLONGED BREATHING OF VAPORS. CONSULT MSDS FOR ADDITIONAL SAFETY INFO.

CAUTION: PROPELLER DEICER REPAIR IS LIMITED TO REFURBISHMENT OF EDGE SEALER. SEE LATEST REVISION OF PROPELLER ELECTRICAL DE-ICE BOOT REMOVAL AND INSTALLATION MANUAL, HARTZELL MANUAL NO. 182 (61-12-82).

GOODRICH PROPELLER DE-ICERS ARE NOT REPAIRABLE. DEICER SURFACE DAMAGE, SUCH AS PUNCTURES, CUTS, SCUFFS, EROSION, THAT EXPOSES THE WIRE OR ETCHED ELEMENT IS GROUNDS FOR REPLACEMENT, AS ADDITION OF REPAIR MATERIAL COULD AFFECT DEICER PERFORMANCE.

CAUTION: DISPOSE OF UNUSED MEK AND OTHER CHEMICALS AND SOLVENTS IN A MANNER CONSISTENT WITH LOCAL LAWS AND/OR ENVIRONMENTAL PROTECTION AGENCY REGULATIONS.

(1) Inspection and Testing

See Inspections, below, and Testing, below.

If tests or inspection show the deicer to have an open circuit, to be the wrong resistance or it has visible damage, remove it and install a new deicer.

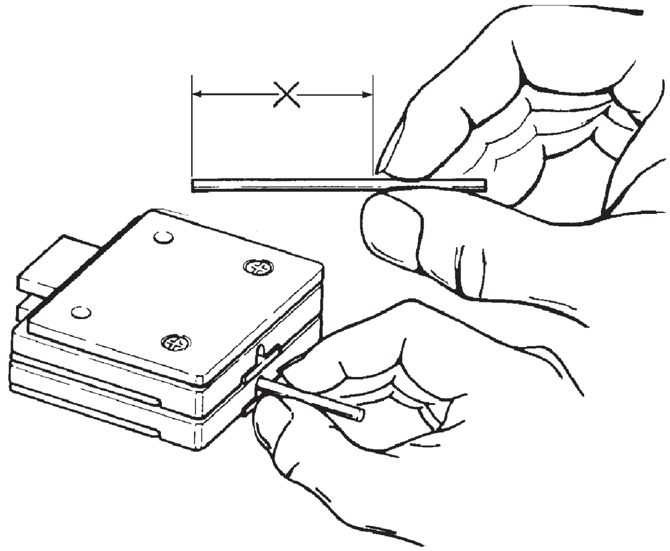
NOTE: Replacement deicers may be ordered from the Goodrich Corporation.

(2) Removal and Installation

NOTE: The following applies to factory propeller installations only.

See latest revision of Hartzell's Propeller Electrical De-ice Boot Removal and Installation Manual, Manual No. 182 (61-12-82).

During measurement only 1/16th inch of brush should be allowed to protrude from brush block. This is the normal position of the brush when installed on the aircraft.



Modular Brush Assemblies	X Dimension Must Replace
Modules having brushes with rods	23/64 IN.
Modules having brushes without rods	1 25/64 IN.

Measuring Brush Assemblies
Figure 4

B. Brush Modules

(1) Replacement

Brush wear may be measured as shown in “Figure 4”. The X - dimension given (in inches) indicates when the brushes must be replaced.

NOTE: Brushes are not offered individually as replacements. When a brush wears out, the module containing it should be replaced.

(a) Remove the modular brush assembly from the aircraft, by removing the attachment hardware, and disconnecting the engine wire harness.

(b) Remove assembly screws and separate modules and spacers.

NOTE: The part number of each module is etched into the surface of the plastic housing; replace with the same part number module.

(c) Restack modules and spacer as shown in “Figure 3”.

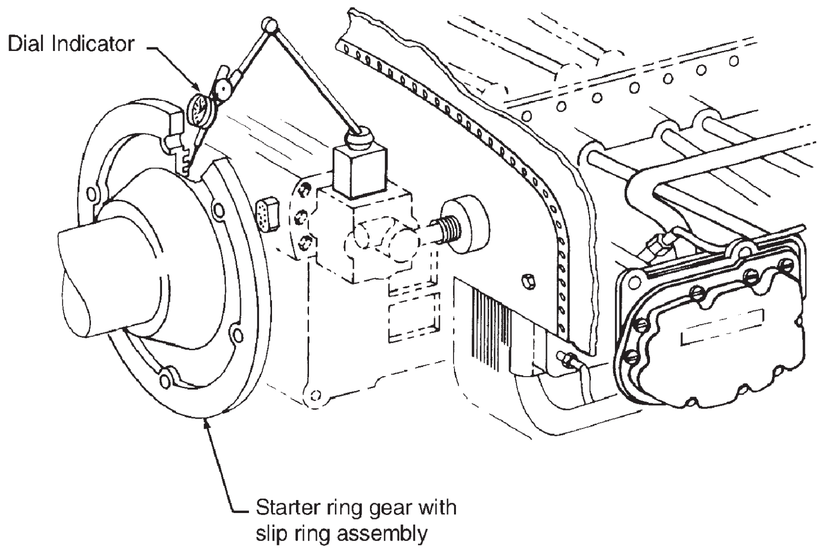
NOTE: Ascertain flat washer is positioned between star washer and housing.

(d) Reconnect aircraft wire harness and ensure adjacent ring terminals are not touching.

(e) Install assembly on aircraft and check adjustment.

(2) Alignment

Any time the brush block assembly is removed or replaced, alignment at reinstallation must be checked as described under Inspections, below, and as shown in “Figure 3”.



Suggested Use of Dial Indicator
Figure 5

C. Metal Oxide Varistor (MOV) Module (S/N's 4636132 and up)

See "Figure 3".

The MOV module must be replaced in the event of a lightning strike.

(1) Disconnect electrical leads, noting their relative position to facilitate reconnection. Loosen and remove screws holding module in position. Remove module.

(2) Position new module, place screws in position, tighten. Connect electrical leads.

D. Slip Ring Assembly

(1) Replacement

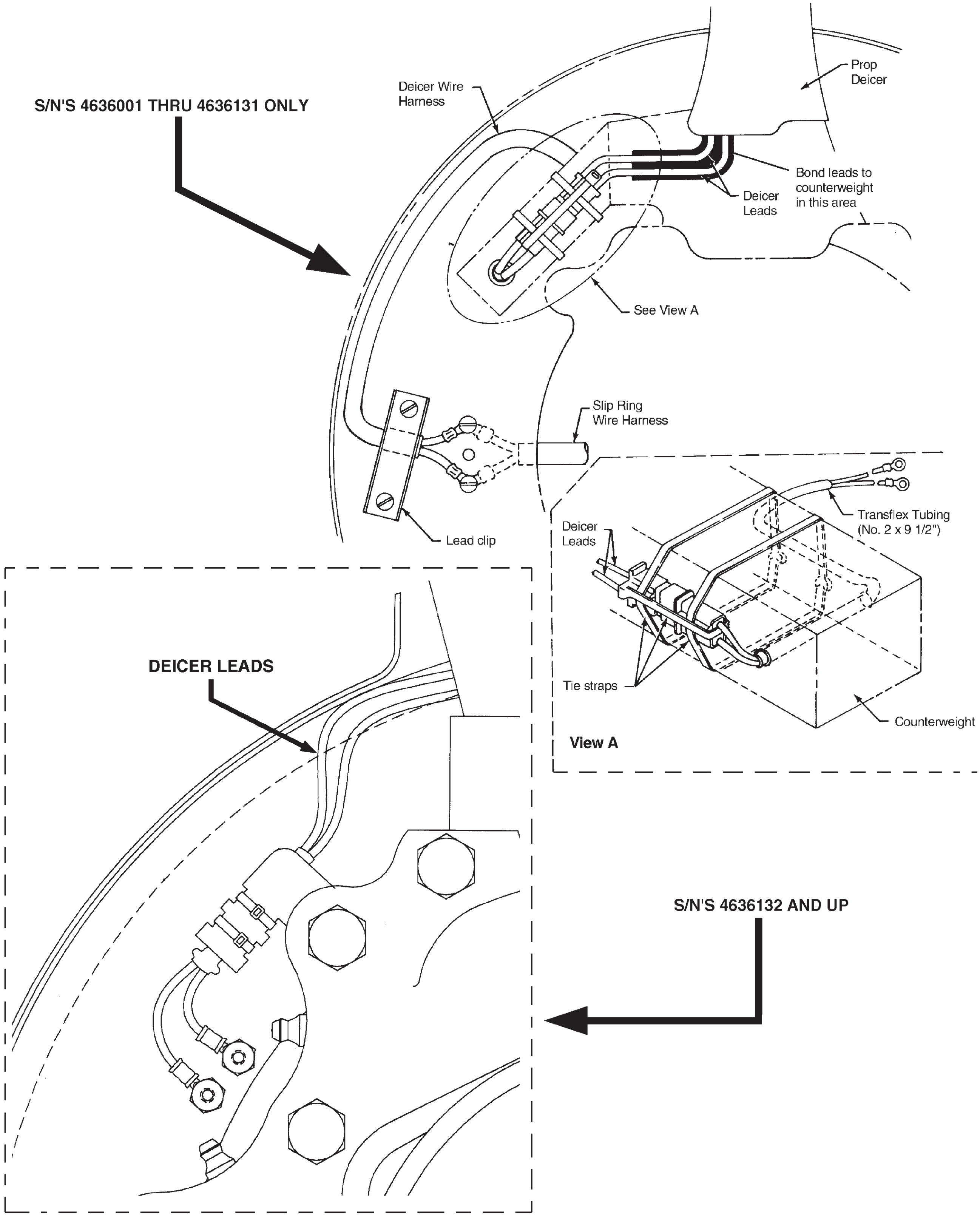
Slip ring assemblies that are open or shorted electrically, cracked or damaged structurally, or which have damaged surfaces beyond the scope of minor repair or clean up, should be replaced.

(2) Alignment

Excessive slip ring run-out will result in severe arcing between the slip ring and brushes and will cause rapid brush wear. If the run-out is not corrected, rapid deterioration of the slip ring and brush contact surfaces will result and lead to eventual failure of the heating system. Check the slip ring run-out with a dial indicator securely attached to the engine with the pointer resting on the slip ring. (See "Figure 5".) Rotate the propeller slowly noting the run-out indicated on the gauge. The total run-out must not exceed 0.005 inch ± 0.0025 inch and 0.002 inch in any 4 inch interval of slip ring travel.

NOTE: Some error may be induced in the readings by pushing in or pulling out on the propeller. Care must be taken to exert a uniform push or pull.

Small amounts of run-out may be corrected by varying the torque on the slip ring mounting bolts between 40 to 100 inch-pounds to obtain the required flatness.



Deicer Wiring Harness Installation
Figure 6

E. Deicer (Heater Boot) Wiring Harness

The propeller deicer wiring harness is secured to the propeller counterweight as follows:

(1) Insert the deicer wire harness thru the 9/32 IN. diameter hole in the prop counterweight.

(2) Connect the plugs of the prop deicers and the deicer wire harness as shown in “Figure 6”.

(3) Install tie strap (P/N MS3367-1-9) between the leads along the length of the plugs. Do not tighten at this time.

(4) Install both tie straps (P/N MS3367-2-9) under the tie strap installed in the previous step and around the counterweight. Do not tighten at this time.

(5) Install transflex tubing over deicer wire harness.

(6) Route transflex tubing under both tie straps (P/N MS3367-2-9) and tighten tie straps.

(7) Tighten the tie strap around the plugs.

(8) Install the terminals on the deicer wire harness.

(9) Install the terminals of the harness to the screws on the spinner bulkhead and tighten the lead clip over the harness.

(10) To assure balance of the propeller assembly, the original balancing weights or their equivalents must be reinstalled. The weights must be left in the original position on the propeller hub. The restrainer and weights should not interfere with any part of the propeller assembly under any condition. If for any reason balance weights were removed, reinstall safety wire on screws. The deicer wire harness must be installed on the propeller as just described.

3. Testing

A. Resistance Test

To determine incorrect resistance, short or open at the brush-to-slip ring contact, disconnect harness at the timer and use low-range ohmmeter to read resistance from each deicer circuit lead (Pins C, D, E and F of harness plug) to ground; it should read 0.58 to 0.67. If this reading is not obtained, disconnect the deicer lead straps to measure deicer resistances individually. Individual deicer should be 1.15 to 1.33. If first check is off limits but second check is satisfactory, trouble is probably in the brush-to-slip ring area; if the second check is off limits, the deicer is damaged and must be replaced.

B. Ammeter

(1) Using the Ammeter

Whether in flight or during ground testing, the ammeter can be used to indicate the general nature of most electrical problems. The troubleshooting chart is primarily based on the use of the ammeter and assumes that the user does understand all normal operating modes of the system.

NOTE: When troubleshooting, first use the “Ammeter Test” on page 306014 and “heat test” (see Inspections - “50 Hour” on page 306015) to determine which circuits are involved. Use circuit diagram for assistance to check voltages or continuity.

(a) Excess current reading on the ammeter always indicates a power lead is shorted to ground. Thus, when trouble of this nature is found, it is vital that the grounded power lead be located and corrected.

(b) A considerable number of timers that have been returned for repair proved to be fully workable when tested. Accomplish the test described under “BFG Timer,” below, before concluding that the timer is defective.

(c) Defective wiring in propeller-mounted components may be indicated by normal current readings in ground checkouts (propeller not rotating) but low current with propeller rotating at cruise RPM.

CHART 2
DEICER ELECTRICAL RESISTANCE

	Maximum	Minimum
Two (2) Blade Propeller		
Single Blade	4.90	4.70
Both Blades in Parallel	2.45	2.35
Three (3) Blade Propeller		
Single Blade	4.56	4.12
All Blades in Parallel	1.52	1.37

(2) Ammeter Test

(a) Check ammeter by connecting calibrated test ammeter or volt-ohmmeter with ammeter scale, into de-icing system circuit between bus bar and system ammeter. Activate system and check amperage reading on test ammeter. Reading should agree with reading on system ammeter. Deactivate system and remove test ammeter or volt-ohmmeter. If readings are different, and deicing system operates satisfactorily, replace system ammeter.

(b) If the timer and ammeter tests are satisfactory, circuit breaker and switch are functioning correctly. If testing is unsatisfactory, and timer and ammeter check out satisfactorily, repair or replace circuit breaker and/or switch.

C. Propeller Deicers (i.e. - Heater Boots)

(1) Final Electrical Check

(a) Verify that deicers are installed so that blades can be moved throughout the entire blade pitch range without placing deicer leads or straps under tension. Make certain that all terminals are tight. Do not over torque.

(b) Check the electrical resistance between terminals or between the slip rings. The reading should be per Chart 2.

(2) Functional Check

(PIR-PPS60054-5 Rev. E., -6 Rev. E., -7. Rev. B.)

(a) S/N's 4636001 thru 4636131 only:

CAUTION: DO NOT OPERATE PROPELLER HEAT EQUIPMENT WITH ENGINE NOT RUNNING FOR MORE THAN 30 SECONDS.

1) Lock brakes, with engine operating at 2,000 rpm, turn PROP HEAT switch ON.

2) De-ice ammeter shall indicate in the green range (8 to 12 amps). Cycle time is 90 seconds ON and 90 seconds OFF.

3) Turn PROP HEAT switch OFF. Turn engine OFF.

(b) S/N's 4636132 and up, S/N's 4692001 and up:

CAUTION: DO NOT OPERATE PROPELLER HEAT EQUIPMENT WITH ENGINE NOT RUNNING FOR MORE THAN 10 SECONDS.

1) Lock brakes, with engine operating at 2,000 rpm, and ALTR NO. 1 and ALTR NO. 2 both ON, turn PROP HEAT switch ON.

2) Verify that the PROP HEAT switch flashes rapidly for 30 seconds, indicating the deicer is in the ON cycle. After approximately 30-45 seconds, the flash rate should reduce, indicating the deicer is in the OFF cycle. The switch will continue to flash at the reduced rate as long as the airplane remains on the ground or until the PROP HEAT switch is deselected. Verify that the PROP DE-ICE FAIL annunciator, or PROP HT FAIL message (on G1000/G1000 NXi equipped aircraft), does not illuminate.

3) Verify that the ship's total amperage (LH and RH combined) shows an increase of approximately 15 amps with the 3-bladed propeller, or 10 amps with the 2-bladed propeller, when the PROP HEAT is cycled ON.

4) Turn PROP HEAT switch OFF. Turn engine OFF.

5) Test PROP DE-ICE FAIL annunciator as specified in Chapter 31.

D. BFG Timer (S/N's 4636001 thru 4636131 only)

Field experience indicates that too often the timer is considered at fault when the true trouble lies elsewhere. Before removing a timer as defective, perform this test:

(1) Disconnect wire harness at timer and, with PROP HEAT switch ON, check voltage from Pin B of harness plug to ground. If system voltage is not present, the fault is not in the timer. If system voltage is present at Pin B, check ground circuit using ohmmeter from Pin G to ground. If no circuit is shown, the fault is in ground lead, not in timer. If ground connection is open, the timer step switch will not change position.

(2) When power and ground circuits have been checked, connect a jumper wire from Pin B ofharness to B contact of timer socket to power timer. Connect a jumper wire from Pin G of harness to G contact of timer socket to complete the power circuit. Now use voltmeter from ground to the timer socket and check that timer is cycling to deliver system voltage to the C contact. Voltage must be delivered for approximately 90 seconds.

(3) If the timer meets these requirements, it is not the cause of the trouble. If it fails to perform as indicated, the trouble does lie in the timer and it should be replaced.

4. Inspections

A. 50 Hour

(1) S/N's 4636001 thru 4636131 only:

(a) Lock brakes and operate engine at near takeoff power. Turn PROP HEAT switch ON and observe deicer ammeter for at least two minutes. Ammeter needle must rest within the shaded band, except for a "flicker," approximately every 90 seconds, as the step switch of the timer operates. Any movement of the needle other than the normal 90 second flicker indicates a short or open that must be located and corrected.

(b) Perform "heat test" as follows: with engine stopped, turn PROP HEAT switch ON and feel deicer on propellers for proper sequence of deicer operation. The deicers should cycle ON for 90 seconds, then OFF for 90 seconds and repeat. Temperature rise should be noticeable and each deicer should warm for about 90 seconds. Local hot spots indicate surface damage of deicers requiring replacement of deicer.

(c) Remove spinner dome and engine cowling. With assistant observing deicer ammeter and with PROP HEAT switch ON, flex all accessible wiring, particularly the deicer wire harness, leads from slip ring assembly and the firewall electrical connectors and their wiring. Any movement of the ammeter needle other than the "90 second flicker" of cycling indicates a short or open that must be located and corrected.

(2) S/N's 4636132 and up, S/N's 4692001 and up:

(a) Lock brakes, with engine operating at 2,000 rpm, and ALTR NO. 1 and ALTR NO. 2 both ON, turn PROP HEAT switch ON.

(b) Verify that the green PROP HEAT switch flashes rapidly for 30 seconds, indicating the deicer is in the ON cycle. After approximately 30-45 seconds, the flash rate should reduce, indicating the deicer is in the OFF cycle. The switch will continue to flash at the reduced rate as long as the airplane remains on the ground or until the PROP HEAT switch is deselected. Verify that the PROP DE-ICE FAIL annunciator, or PROP HT FAIL message (on G1000/ G1000 NXi equipped aircraft), does not illuminate.

(c) Verify that the ship's total amperage (LH and RH combined) shows an increase of approximately 15 amps with the 3-bladed propeller, or 10 amps with the 2-bladed propeller, when the PROP HEAT is cycled ON.

(d) Turn PROP HEAT switch OFF. Turn engine OFF.

(e) Perform "heat test" as follows: with engine stopped, turn PROP HEAT switch ON and within 30 seconds feel deicer on propellers to verify ON cycle. Within 30-45 seconds of switching ON, the system will shut itself OFF - verify that this occurs.Temperature rise should be noticeable. Local hot spots indicate surface damage of deicers requiring replacement of deicer.

B. 100 Hour

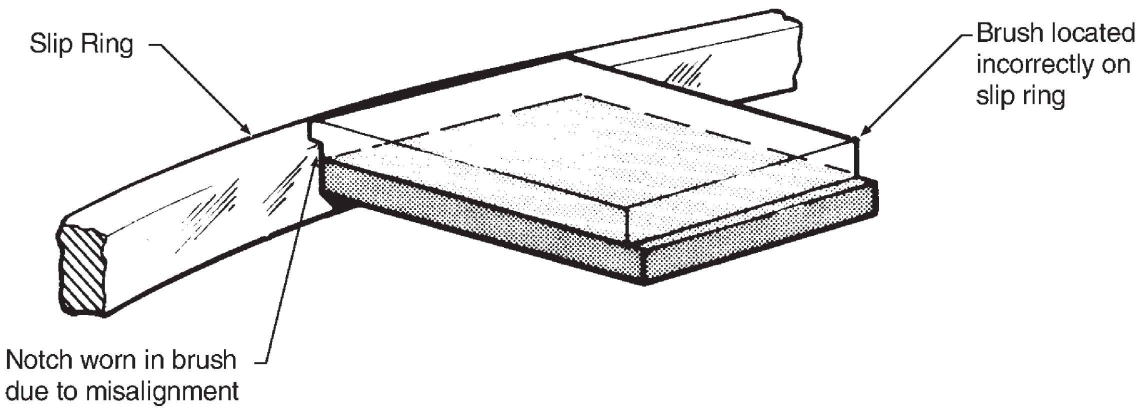
(1) Remove cowling. Conduct appropriate 50 Hour Inspection, above.

(2) Check for radio noise or radio compass interference by operating the engine at near takeoff power with radio gear ON while turning PROP HEAT switch ON and OFF. If noise or interference occurs with PROP HEAT switch ON and disappears when switch is OFF, refer to troubleshooting chart.

(3) Ascertain that all clamps, clips, mountings and electrical connections are tight. Check for loose, broken or missing safety wire.

(4) Deicers: Closely check deicers for wrinkled, loose or torn areas, particularly around the outboard end. Check that terminals and terminal studs are not shorting out to each other or the propeller hub. Inspect wire harness for evidence of chafing or shorting. Ensure that all clamps and tie straps are properly installed. Look for abrasion or cuts, especially along the leading edge and the flat or thrust face. If deicer wires are exposed in damaged areas or if rubber is found to be tacky, swollen or deteriorated (as from oil or solvent contact), replace the damaged deicer (see Deicers (Propeller Deice System), below).

NOTE: Operate propeller from full low pitch (high RPM) to feathering and check that deicer lead wires do not come under tension or are pinched by propeller blade (refer to "Figure 3" on page 30607. and "Figure 6" on page 306012).



Centering Brushes on Slip Rings
Figure 7

(5) Slip Rings: Check slip rings for gouges, roughened surface, cracks, burned or discolored areas, and for deposits of oil, grease or dirt.

CAUTION: DISPOSE OF UNUSED MEK AND OTHER CHEMICALS AND SOLVENTS IN A MANNER CONSISTENT WITH LOCAL LAWS AND/OR ENVIRONMENTAL PROTECTION AGENCY REGULATIONS.

(a) Clean greasy or contaminated slip rings with CRC 2-26 solvent or with Stoddard solvent. If Stoddard solvent is used, wipe off with a clean cloth dampened with MEK to avoid leaving a film.

(b) If uneven wear is found or if wobble is noticed, set up a dial indicator as shown in "Figure 5" on page 306011 and check alignment of the slip rings to the propeller shaft as explained in this section.

(6) Modular Brush Assemblies - Brushes: Examine mounting brackets and housing for cracks, deformation or other physical damage.

(a) Test that each brush rides fully on its slip ring over 360°. "Figure 7" shows the wear pattern if this condition is not corrected. If alignment is off, shim where brush block is mounted to bracket or adjust mounting bracket support arm.

NOTE: The shim is a series of laminates and may be peeled for proper alignment of brushes to slip ring

(b) Check for proper clearance of brush block to slip rings (see "Figure 3" on page 30607). Loosen mounting screws and move in elongated holes to correct block position and tighten securely.

(c) Visually check brush block for 2° angle of attack (see "Figure 3"). If not, loosen mounting screws and twist block, but be sure to hold clearance limits shown when tightening.

1) Note direction of slip ring rotation.

2) Place template against face of brush assembly.

3) Slide brush assembly and remove template.

(7) System Wiring

Visually inspect wiring from brush blocks through firewall, to timer, to ammeter, to switch and to aircraft power supply for breaks or damage. In such case, check continuity through affected harness. Use the wiring diagram to trace circuitry.

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CHAPTER

31

INDICATING / RECORDING SYSTEMS

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CHAPTER 31

LIST OF EFFECTIVE PAGES

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CHAPTER 31 - INDICATING / RECORDING SYSTEMS

TABLE OF CONTENTS

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INDEPENDENT INSTRUMENTS

CO Pulse Oximeter and Carbon Monoxide Detector

S/N's 4636633,4636652 and up.

The Model 455 Pulse Oximeter and Carbon Monoxide Detector is designed to detect, measure, and provide a visual alert to the crew before the cockpit level of carbon monoxide (CO) reaches a critical level. Additionaly, the occupants of the aircraft are able to monitor their physiological condition using a pulse oximeter installed in the instrument panel. It measures SPO_2 (oxygen saturation percentage in blood) and heart rate (see "Figure 1" and "Figure 2" on page 31202). CO PPM is displayed on the engine page of the MFD (see "Figure 3" on page 31203).

A. Troubleshooting

CO Detectors must be returned to manufacturer for repair, calibration or overhaul.

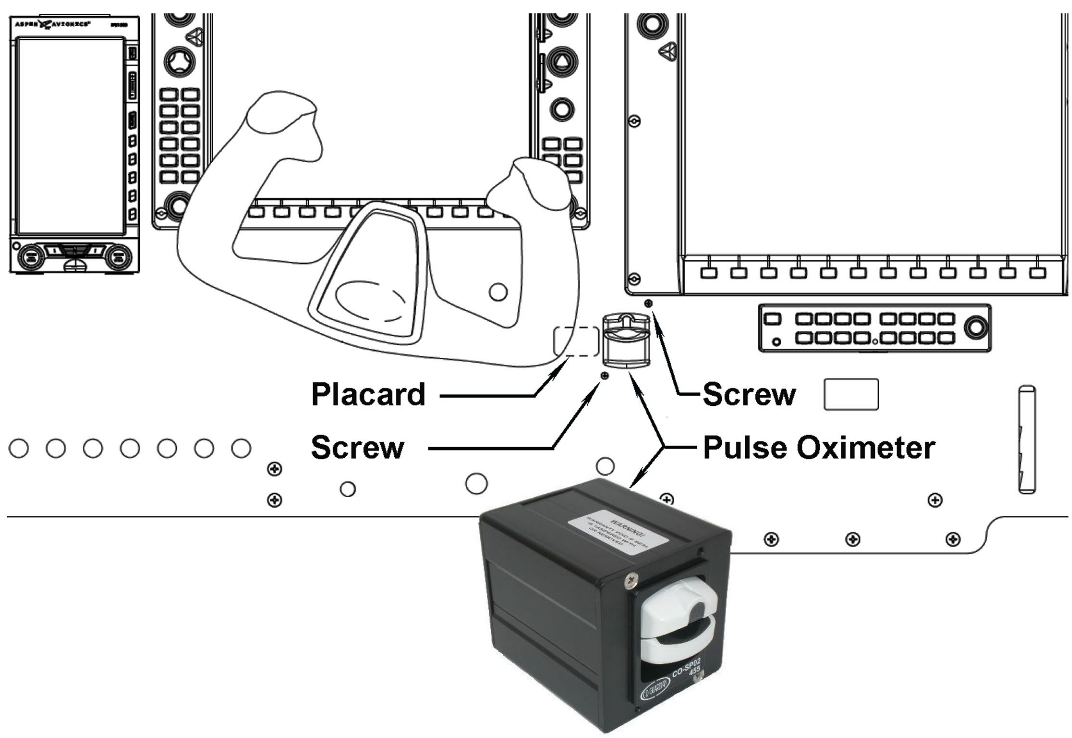
B. Replacment Interval

The sensor life is five (5) years from date of installation.

C. Inspection/Maintenance

The carbon monoxide detector and associated equipment do not require periodic scheduled servicing or periodic scheduled preventive maintenance. Conduct a general visual inspection for system integrity, installation security, corrosion and chaffing.

NOTE: Be sure the vent on the faceplate is free of obstructions. Any failures of the system are evident to the pilot through a system failure message on the PFD.



Pulse Oximeter
Figure 1

D. Removal

(1) Ensure aircraft power is off.

(2) Remove two (2) mounting screws and carefully pull the pulse oximeter away from the instrument panel to allow access to the electrical connector on the rear.

(3) Remove the electrical connector.

E. Installation

(1) Attach the electrical connector at the rear of the pulse oximeter.

(2) Feed the elctrical cable through the mounting hole in the instrument panel and place the pulse oximeter in the instrument panel.

(3) Secure the pulse oximeter using the two (2) mounting screws.

F. Self Test Sequence At Startup

(1) When the airplane master battery switch is selected ON, the 455 Detector goes through a self-test routine.

(2) The self-test checks for functionality of critical components such as the CO sensor, temperature sensor, pressure sensor, pulse oximeter, and integrity of the system and remote display will remain off if everything working properly. The RS232 MFD will show no CO on the CO Detector page.

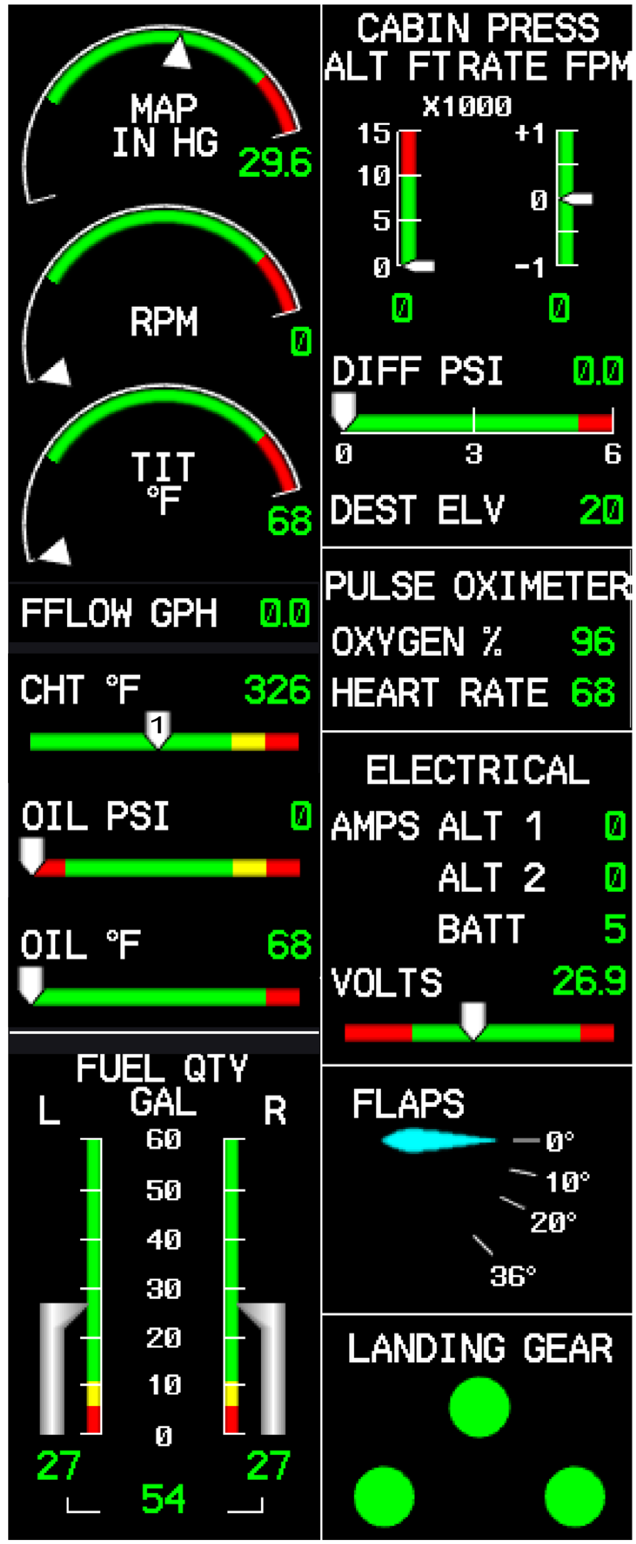
G. Alarm Indicator

WARNING: ANY FAILURE OF THE SYSTEM WILL SHOW A SYSTEM FAILURE MESSAGE ON THE PFD. RETURN THE UNIT TO CO GUARDIAN FOR REPAIR OR REPLACEMENT.

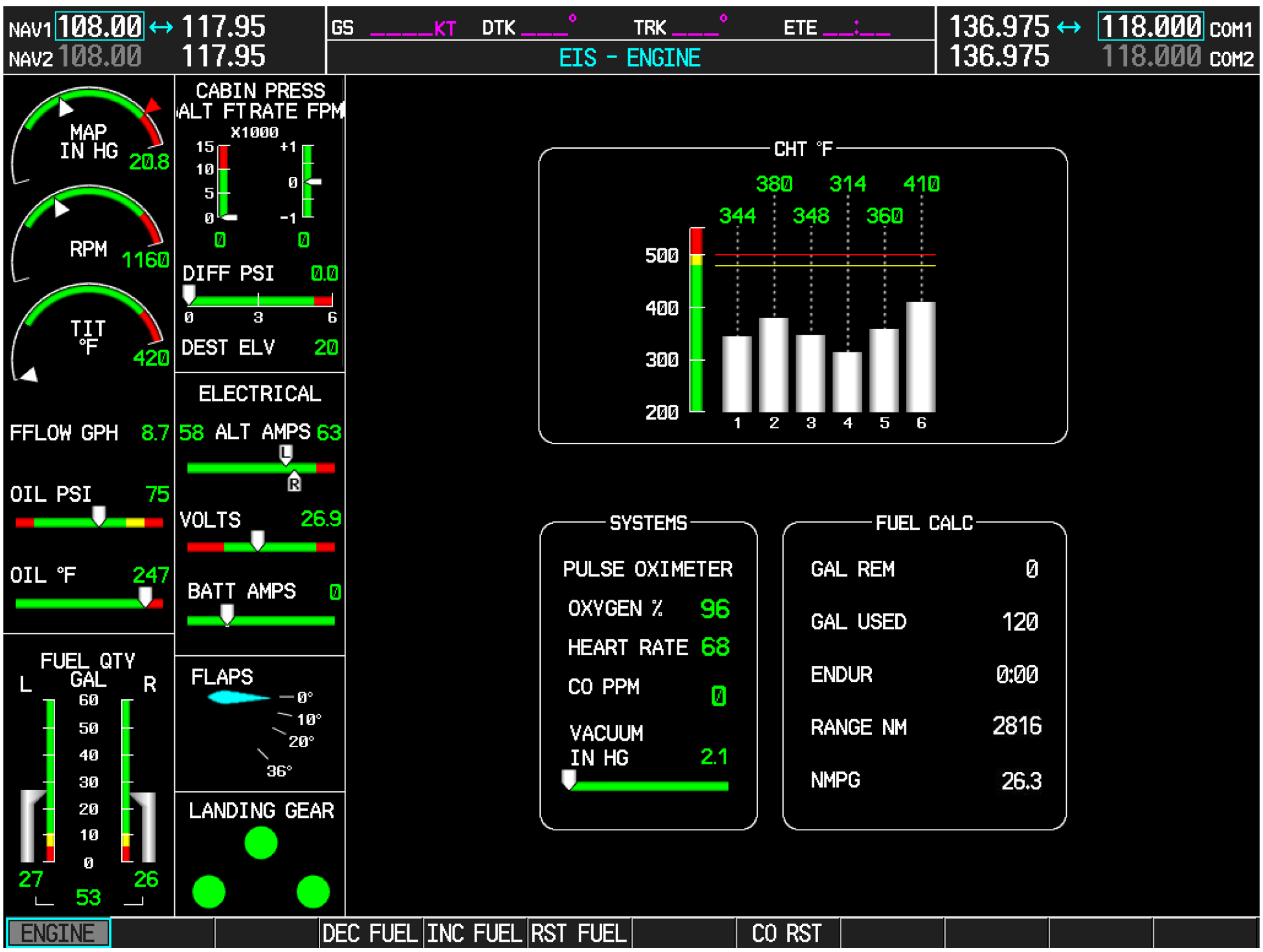
(1) Relevant alert messages will display on the Primary Flight Display (PFD).

NOTE: The alert message will stay on until the CO level goes below 50 parts per million (PPM) by volume of carbon monoxide concentration.

(2) The CO ALERT can be reset on the engine page of the MFD (see “Figure 3”).



MFD EIS Strip
Figure 2



MFD Engine Page
Figure 3

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CENTRAL WARNING SYSTEMS

1. Annunciator Panel

In conventional gauge or Avidyne Entegra equipped aircraft, an annunciator panel is located in the upper center section of the instrument panel and consists of a group of warning lights as follows: (shown with optional equipment).

NOTE: Refer to Annunciator Panel figures to determine specific annumciator usage.

NOTE: Crew Alerting System (CAS) messages are displayed on the MFD when Garmin G1000/ G1000 NXi is installed.

ALTERNATOR NO. 1 INOP ALTERNATOR NO. 2 INOP	When either alternator fails or is selected OFF, the appropriate annunciator light will illuminate.
ANNUNCIATOR INOP	Illuminates if the annunciator fail relay switch is faulty and/or the ANNUN circuit breaker trips.
BOOST PUMP	Illuminates when the fuel boost pump in the tank being used fails to generate sufficient pressure. In normal operation, will also illuminate briefly when switching tanks. Neither BOOST PUMP will function when the fuel selector is set to OFF or is positioned between the LEFT and RIGHT detents.
CABIN ALTITUDE	Illuminates if cabin pressure altitude exceeds 10,000 feet.
DOOR AJAR	Illuminates if either the forward baggage door or the cabin door is not closed and latched.
FLAPS	Illuminates when the flap motor protection circuit breaker trips.
FUEL PRESS	Illuminates when the engine-driven fuel pump fails to generate sufficient pressure.
FUEL IMBALANCE	S/N's 4636021 and up. Illuminates if a fuel quantity difference of greater than 10 gallons is detected for more than 60 seconds between the left and right tanks. Remains lighted until the imbalance is corrected.
GEAR WARN	On the ground - illuminates when the landing gear selector is placed in the UP position with the aircraft weight still on the gear. In the air - illuminates when the gear is in transit; when all three gear are not fully down and locked, or when the gear is not fully UP but the gear doors are closed; when the throttle is reduced below 14 inches of manifold pressure and the landing gear are not in the DOWN position; or if the flaps are extended more than 10° and the landing gear are not in the DOWN position.
HYDRAULIC PUMP	Indicates the hydraulic pump is being supplied with electrical power. Normally illuminated only during landing gear operation.
ICE DETECT FAIL	S/N's 4636132 thru 4636313 only. Illuminates if a fault develops in either the ice detection sensor or controller.
LOW BUS VOLTAGE/VOLTS	Illuminates when system voltage drops below 25 ± 0.3 Vdc. Normally on when just the airplane's battery or an external 28 Vdc battery is used for electrical power.
OIL PRESS	Illuminates if oil pressure is below 24 PSI. As oil pressure drops, there may be as much as an 8 PSI lag (dead band) before the light comes on.

OXYGEN	Illuminates whenever one of the oxygen generators has been activated. Remains lighted until the expended oxygen generator(s) is/are replaced.
PITOT HEAT OFF/INOP	S/N's 4636021 thru 4636374, S/N's 4692001 and up. Self-explanatory.
PITOT HEAT OFF	S/N's 4636375 and up. Self-explanatory.
L PITOT HEAT R PITOT HEAT	S/N's 4636375 and up. Illuminates when the appropriate pitot system fails.
PROP DE-ICE FAIL	S/N's 4636132 and up only. Illuminates if a fault develops in the PROP HEAT system.
SELECT DE-ICE	S/N's 4636132 thru 4636313 only. Illuminates when the SMARTboot™ Ice Detection system calculates the ice buildup on the vertical fin boot in excess of 1/4 inch. Will remain illuminated until at least 90% of the ice cap has been removed from the right horizontal stabilizer boot. This annunciator is advisory only.
STALL WARN FAIL	Illuminates if the lift computer fails and/or the STALL WARN circuit breaker trips.
START(ER) ENGAGE	Illumination during engine cranking is normal. Illumination at any other time indicates a problem.
SURFACE DE-ICE	Illuminates when the de-ice boots inflate, i.e. - when the SURFACE DE-ICE switch is depressed and the pneumatic de-ice system cycles in sequence: tail, lower wing, upper wing.
VACUUM NO. 1 INOP VACUUM NO. 2 INOP	S/N's 4636021 and up. If either vacuum pump fails, the appropriate annunciator light will illuminate.
WINDSHIELD HEAT FAIL	Illuminates when an overtemperature condition exists or the windshield temperature sensor fails. In normal operation, may also illuminate briefly when switching from HIGH to LOW windshield heat.

ALTERNATOR NO. 1 INOP	BOOST PUMP	FUEL PRESS	LOW BUS VOLTAGE	CABIN ALTITUDE	STALL WARN FAIL
ALTERNATOR NO. 2 INOP	GEAR WARN	DOOR AJAR	FLAPS	STARTER ENGAGE	WINDSHIELD HEAT FAIL
OXYGEN	———	OIL PRESS	HYDRAULIC PUMP	SURFACE DE-ICE	ANNUNCIATOR INOP

Annunciator Panel
Figure 1

Effectivity
4636001 thru 4636020

VACUUM NO. 1 INOP	VACUUM NO. 2 INOP	———	———	———	FUEL IMBALANCE
ALTERNATOR NO. 1 INOP	BOOST PUMP	FUEL PRESS	LOW BUS VOLTAGE	CABIN ALTITUDE	STALL WARN FAIL
ALTERNATOR NO. 2 INOP	GEAR WARN	DOOR AJAR	FLAPS	STARTER ENGAGE	WINDSHIELD HEAT FAIL
OXYGEN	PITOT HEAT OFF / INOP	OIL PRESS	HYDRAULIC PUMP	SURFACE DE-ICE	ANNUNCIATOR INOP

Annunciator Panel
Figure 2

Effectivity
4636021 thru 4636131

VACUUM NO. 1 INOP	VACUUM NO. 2 INOP	OIL PRESS	HYDRAULIC PUMP	PITOT HEAT OFF / INOP	FUEL IMBALANCE
ALTERNATOR NO. 1 INOP	BOOST PUMP	FUEL PRESS	LOW BUS VOLTAGE	CABIN ALTITUDE	STALL WARN FAIL
ALTERNATOR NO. 2 INOP	GEAR WARN	DOOR AJAR	FLAPS	STARTER ENGAGE	WINDSHIELD HEAT FAIL
PROP DE-ICE FAIL	OXYGEN	SELECT DE-ICE	ANNUNCIATOR INOP	ICE DETECT FAIL	SURFACE DE-ICE

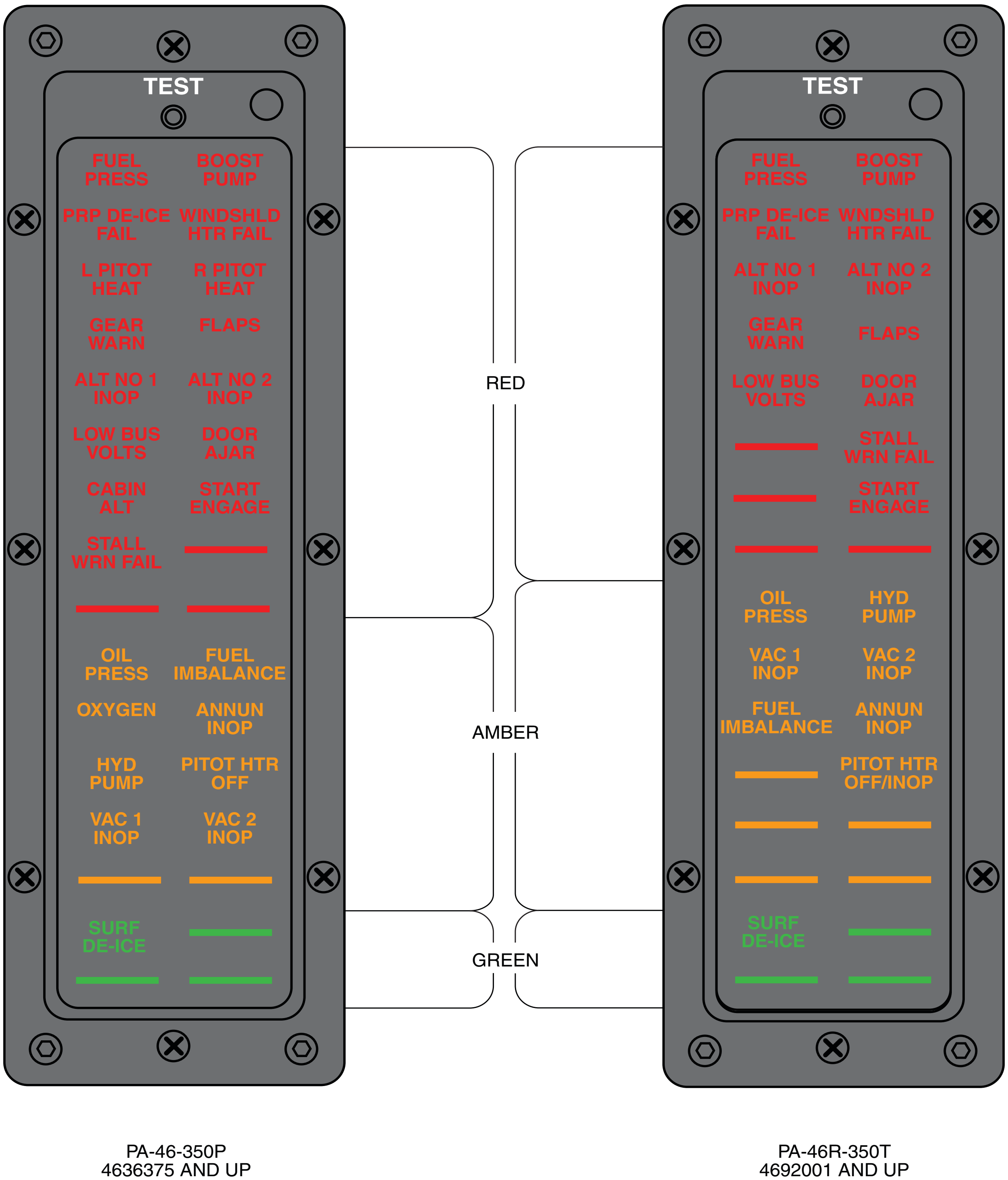
Effectivity
4636132 thru 4636313,
less 4636299

Annunciator Panel
Figure 3

VACUUM NO 1 INOP	VACUUM NO 2 INOP	OIL PRESS	HYDRALIC PUMP	PITOT HEAT OFF/INOP	FUEL IMABLANCE
ALTERNATOR NO 1 INOP	BOOST PUMP	FUEL PRESS	LOW BUS VOLTAGE	CABIN ALTITUDE	STALL WARN FAIL
ALTERNATOR NO 2 INOP	GEAR WARN	DOOR AJAR	FLAPS	STARTER ENGAGE	WINDSHIELD HEAT FAIL
PROP DE-ICE FAIL	OXYGEN	———	ANNUNC INOP	———	SURFACE DE-ICE

Effectivity
4636314 thru 4636374,
and 4636299

Annunciator Panel
Figure 4



Effectivity
4636375 and up, Avidyne
4692001 and up, Avidyne

Annunciator Panel
Figure 5

2. Annunciator Tests

The sequence of the following tests may be varied.

A. With the fuel selector OFF and the emergency fuel pump OFF, turn ON the Battery Master switch and verify that the alternator #1 inop, alternator #2 inop, boost pump, fuel pressure, low bus voltage/volts, oil pressure, vacuum #1 inop, vacuum #2 inop, left, nose and right landing gear annunciator lights are illuminated.

NOTE: Gear annunciators are separate from the annunciator panel assembly.

B. Press the annunciator test button. Check that all annunciators are illuminated. Locate the day/night switch and activate the switch to both day and night positions with the annunciator test button pressed. Check that all annunciator and gear lamps illuminate bright for day and dim for night.

NOTE: In PA-46-350P S/N's 4636132 and up; Annunciator Inop does not dim for night.

C. Function test the gear warning and landing gear lamps per 32-30-00.

D. Verify that the door ajar lamp illuminates when the upper or lower portion of the main cabin door or the baggage door is unlatched, and extinguishes when all doors are closed and latched.

E. Momentarily active the fuel pump switch. Check that the fuel pressure annunciator extinguishes then illuminates when the switch is turned OFF. Momentarily turn the fuel selector to left and right positions. Check that the boost pump annunciator extinguishes in both positions then illuminates when the selector is turned OFF.

F. In PA-46-350P aircraft, if the optional oxygen generator system is installed, press the annunciator test button to verify that the oxygen annunciator illuminates.

NOTE: This test only verifies that the annunciator lamps are operational. The functional oxygen system check is conducted per requirements of the drawing.

G. Start the engine and observe that the start(er) engage annunciator illuminates during the time the starter is engaged. Observe that the alternator #1 inop, alternator #2 inop, fuel pressure, low bus voltage/volts, oil pressure, vacuum #1 inop and vacuum #2 inop light extinguishes as the engine starts. Note that alternator #1, alternator #2, fuel pressure, low bus voltage, oil pressure and vacuum outputs are normal.

NOTE: Vacuum annunciation only available on aircraft equipped with the optional de-ice system.

H. During engine start, observe the oil pressure gage and annunciator light. The annunciator light should illuminate when oil pressure is 15 pounds per square inch (psi) or less, and extinguish above 15 psi.

I. With the engine at idle RPM, place the alternator switches in the OFF position. Check that the alternator annunciators illuminate and correspond with the appropriate switch.

J. If the surface de-ice installation is installed, check that the surface de-ice annunciator illuminates when the surface de-ice switch is activated.

K. Shut the engine down and, if equipped with the optional de-ice system, verify that the vacuum low annunciator lights at 3.8 inches of vacuum. Check that other required annunciator lights are ON.

L. In PA-46-350P aircraft, press the annunciator test button and verify that the cabin altitude annunciator lights.

NOTE: Flight test is required to check functional operation of cabin altitude annunciator. Cabin altitude annunciator should illuminate at 10,000 + 500 feet.

M. If the propeller de-ice system is installed, function check the annunciator lamps per 30-60-00.

N. Function check the flaps annunciator lights/message per 27-50-00.

3. Annunciator Removal and Installation

   See 39-10-00.

4. Lamp Replacement

   See 33-10-00.

5. Wiring

   See 33-10-00 and the schematics presented in 91-31-50.

6. Annunciation in Garmin G1000/G1000 NXi-equipped Airplanes

Annunciation is integrated into G1000/G1000 NXi system with the exception of landing gear warning in the Mirage and Matrix. The Crew Alerting System (CAS) consists of Master Warning and Master Caution Indicators operating in conjunction with CAS text messages. CAS text messages appear in the lower left area of the MFD during normal operations and in the right side area of the PFD's during reversionary mode operation. The Master Warning and Master Caution Indicators are illuminated push-button switches centered above the pilot's PFD in the Mirage and Matrix. In the M350 the Master Warning and Master Caution Indicators are integrated into the G1000/G1000 NXi displays. The severity of CAS messages are categorized as Warning, Caution and Advisory as follows:

A. Warning (**red on black**) – Immediate crew awareness and action required; Master Warning triggered Warnings are accompanied by a continuous aural chime.

B. Caution (**amber on black**) – Immediate crew awareness and possible future corrective action required; Master Caution triggered Cautions are usually accompanied by a single aural chime of approximately 1.5 seconds in duration.

C. Advisory (**white**) – Advisory information; and are not accompanied by aural chimes. no immediate action required.

NOTE: Chart 1 is an example only. For detailed listings see the appropriate Garmin Pilot's Guide and the Garmin Cockpit Reference Guide. For instance, M350 landing gear warnings are not shown in Chart 1.

CHART 1 (Sheet 1 of 3)
CREW ALERTING SYSTEM

— WARNINGS - RED - REPEATING AURAL CHIME —

Message	Cause
ALTR 1 FAIL	ALTR NO. 1 switch selected ON and the alternator has failed.
ALTR 2 FAIL	ALTR NO. 2 switch selected ON and the alternator has failed.
CABIN ALT 10000	Cabin altitude is greater than 10,000 feet.
FUEL PRESS LOW	Fuel pressure is below approximately 10 psig.
DOOR AJAR	Cabin door or nose baggage door is not properly closed and latched while the engine is running.
L FUEL QTY LOW	Left tank fuel quantity is less than or equal to 5.0 gallons while the engine is running.
R FUEL QTY LOW	Right tank fuel quantity is less than or equal to 5.0 gallons while the engine is running.
SPEEDBRAKES EXTD	Speedbrakes are extended during the takeoff phase of flight.
WNDSHLD OVERTEMP	Windshield temperature exceeds 170°F or the windshield temperature sensor has failed.

CHART 1 (Sheet 2 of 3)
CREW ALERTING SYSTEM

— CAUTIONS - AMBER - SINGLE AURAL CHIME —

Message	Cause
BOOST PUMP FAIL	Fuel boost pump in the tank being used fails to generate sufficient pressure. This message is displayed briefly when switching tanks.
FLAP FAIL	Flap system failure due to an overcurrent condition in the flap motor/actuator circuit.
FUEL IMBALANCE	Fuel quantity imbalance greater than 10 gallons while the engine is running.
HYDR PUMP ON	Hydraulic pump has been running for 1-second (on ground) or 20 seconds (in flight).
OXYGEN GEN ON	Oxygen generator for one or more of the passengers has been activated or is expended.
L PITOT HT FAIL	Left pitot heat has failed.
R PITOT HT FAIL	Right pitot heat has failed.
PITOT HT OFF	Pitot heat is turned OFF. No aural chime accompanies this action.
PROP HT FAIL	A propeller heat fault has developed or the current is less than 16.0 amps.
SPEEDBRAKES EXTD	Speedbrakes are extended in flight while the engine is OFF, being started or during landing.
STALL WARN FAIL	The stall warning lift computer and/or lift transducer has failed.
STARTER ENGAGED	The starter contactor is engaged for greater than 30 seconds during engine start or anytime while the engine is running.
SURF DE-ICE FAIL	Surface de-ice system has failed to reach appropriate pressure.

CHART 1 (Sheet 3 of 3)
CREW ALERTING SYSTEM

— ADVISORIES - WHITE - NO AURAL CHIME —

Message	Cause
DOOR AJAR	Cabin door or nose baggage door is not properly closed and latched while the engine is OFF.
EMERG FUEL OFF	Emergency fuel pump is OFF while the engine is running and fuel pressure has dropped below 10 psi.
EMERG FUEL ON	Emergency fuel pump is ON after completing the take-off phase of flight.
FUEL IMBALANCE	Fuel quantity imbalance greater than 10 gallons while the engine is OFF.
L FUEL QTY LOW	Left tank fuel quantity is less than or equal to 5.0 gallons while the engine is OFF.
R FUEL QTY LOW	Right tank fuel quantity is less than or equal to 5.0 gallons while the engine is OFF.
SPEEDBRAKES EXTD	Speedbrakes are extended while the airplane is on the ground or after completing the take-off phase of flight.
SURF DE-ICE ON	Surface de-ice system is selected ON and power is being applied to the SURF DE-ICE timer.
VACUUM 1 FAIL	Vacuum No. 1 pressure is less than approximately 2.0 in. of Hg.
VACUUM 2 FAIL	Vacuum No. 2 pressure is less than approximately 2.0 in. of Hg.

7. Aural Alerts

Aural alerts (both voice and chime) are provided to alert the crew and call for their attention to various issues. For detailed listings see the appropriate Garmin Pilot's Guide and the Garmin Cockpit Reference Guide.

CHAPTER

32

LANDING GEAR

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CHAPTER 32

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
32-List of Effective Pages	1	Mar 25/21
	2	Feb 21/20
32-Table of Contents	1	Mar 25/21
	2	Feb 21/20
	3	Feb 21/20
	4	Feb 21/20
32-00-00	1	Nov 30/17
	2	Nov 30/17
32-10-00	1	Sep 15/09
	2	Nov 30/17
	3	Dec 15/14
	4	Dec 15/14
	5	Nov 30/17
	6	Jul 1/10
	7	Nov 30/17
	8	Jul 1/10
	9	Nov 30/17
	10	Sep 15/09
32-20-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Nov 30/17
	5	Feb 15/14
	6	Mar 25/21
	7	Nov 30/17
	8	Mar 25/21
	9	Mar 25/21
	10	Mar 25/21
	11	Mar 25/21
	12	Mar 25/21
	13	Mar 25/21
	14	Mar 25/21
	15	Mar 25/21
	16	Mar 25/21
	17	Mar 25/21
	18	Mar 25/21
	19	Mar 25/21
	20	Mar 25/21v
32-30-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Dec 15/14
	5	Dec 15/14
	6	Feb 21/20
	7	Feb 21/20
	8	Dec 15/14
	9	Feb 21/20
	10	Nov 30/17
	11	Mar 25/21
	12	Dec 15/14
	13	Dec 15/14
	14	Dec 15/14
	15	Mar 25/21
	16	Mar 25/21
	17	Mar 25/21
	18	Nov 30/17
	19	Nov 30/17
	20	Nov 30/17
	21	Nov 30/17
	22	Mar 25/21
	23	Mar 25/21
	24	Nov 30/17
	25	Nov 30/17
	26	Nov 30/17
	27	Nov 30/17
	28	Nov 30/17
	29	Nov 30/17
	30	Nov 30/17
	31	Nov 30/17
	32	Nov 30/17
	33	Nov 30/17
	34	Feb 21/20
	35	Feb 21/20
	36	Feb 21/20
	37	Feb 21/20
	38	Feb 21/20

CHAPTER 32

LIST OF EFFECTIVE PAGES (continued)

CHAPTER 32 - LANDING GEAR

TABLE OF CONTENTS

CHAPTER 32 - LANDING GEAR

TABLE OF CONTENTS (continued)

CHAPTER 32 - LANDING GEAR

TABLE OF CONTENTS (continued)

CHAPTER 32 - LANDING GEAR

TABLE OF CONTENTS (continued)

GENERAL

This chapter provides instructions for the overhaul, inspection and adjustment of the various components of the landing gear and brake systems. Also included are adjustments for the electrical limit, safety and warning switches.

1. Description and Operation

This airplane is equipped with a retractable, tricycle, air-oil strut-type landing gear which is hydraulically operated by an electric hydraulic pump. A selector handle or switch on the instrument panel to the left of the control quadrant is used to select gear UP or DOWN positions.

- In PA-46-350P S/N's 4636001–4636459, 4636461–4636462, and 4636481; and, PA-46R-350T S/N's 4692001–4692133, 4692141, 4692149, and 4692153:

  Gear position is indicated by a display of three green lights (for gear down and locked) located above the gear selector. A red warning light (for gear-in-transit or gear-unsafe conditions) is in the annunciator panel (see 31-50-00).

- In PA-46-350P S/N's 4636460, 4636463–4636480, 4636482–4636632, 4636634–4636651; and, PA-46R-350T S/N's 4692134–4692140, 4692142–4692148, 4692150–4692152, 4692154–4692214:

  Gear position is indicated by a display of three green lights (for gear down and locked) located above the gear selector. A red warning light (for gear-in-transit or gear-unsafe conditions) is located above the three green lights.

- In PA-46-350P S/N's 4636633, 4636652 and up:

  Gear position is indicated by a display of three symbols displayed in the Garmin G1000/G1000 NXi EIS strip on the MFD or PFD. See "Figure 2" on page 323012 and "Figure 3" on page 323013 through "Figure 6" on page 323014 for examples of these symbols at various gear positions.

- There is no separate light to indicate that all gear are fully retracted other than all gear lights / symbols being out.

The landing gear swings to the down position and each actuator extends to its locked position, a switch located on each actuator activates to indicate by a green light that the individual gear is safely down and locked. When the gear begins to retract and the hydraulic actuators unlock, the down limit switch actuates to the NC circuit and, in series with the NC circuit of the up limit switch, allows the gear unsafe light to come on. The gear unsafe light will remain on until the gear is up and all up limit switches are actuated to their normally open (NO) circuit.

A gear warning horn will sound when power is decreased to 15 ± 1 In. Hg. manifold pressure unless all three gear are down. This horn will also sound if the flaps are extended beyond approach position if the gear are not down and locked.

The red gear unsafe light also operates simultaneously with the warning horn. Their primary purpose is to give warning when power is reduced below 15 ± 1 In. Hg. manifold pressure and the landing gear has not reached the down and locked position. This circuit is controlled by three (3) paralleling down limit switches connected in series with a throttle switch located in the control quadrant (see 91-32-60 for electrical schematic).

Each landing gear is retracted and extended by a single hydraulic locking cylinder attached to the oleo strut housing. As the gear retracts, doors enclose each gear through mechanical linkage. The gear are held in their up position by hydraulic pressure on the cylinder.

A hydraulic system modification and power pak upgrade provides a modification to the hydraulic power pak electrical system which will, under certain conditions, bypass the low pressure switch in the event the switch fails to operate.

The gear system has a free-fall secondary extension system which consists of a valve that is controlled from the panel. A downspring is incorporated in the nose gear to overcome air loads resisting gear extension.

The nose gear is steerable through a 60 degree arc by use of the rudder pedals. As the gear retracts, the steering linkage becomes separated from the gear so that rudder pedal action with the gear retracted is not impeded by the nose gear operation. The nose gear rotates 90° to stow in the horizontal position underneath the luggage compartment floor. A steering bungee is also incorporated in the nose wheel steering mechanism.

The main wheels are equipped with dual-piston, single-slotted disc hydraulic brakes.

The parking brake is set by depressing both toe brakes and setting a push-pull valve lock on the instrument panel.

2. Special Tools

These tools are used at the factory and are recommended where rigging, alignment, or installation procedures specify “use a suitable tool.” Order them through any Piper Dealer.

Tool (Tool Number)	**Piper P/N**
Rudder Pedal Neutral Alignment Tool (HF 622724)	762-151
Nose Wheel Centering (i.e. - Neutral) Tool (HF 622722)	762-147
Nose Gear Steering Bellcrank Centering (i.e. - Neutral) Tool (TL 622723)	762-148
Nose Gear Steering Horn / Arm Gap Tool (TL 690672)	N/A
Nose Gear Actuator Alignment Tool (TL 690672)	762-137
Nose Gear Downspring Disassembly/Reassembly Tool (SK VB1152)	762-155
Nose Gear Downspring Remove/Install Tool (SK VB1153) (See also 32-30-00, Figure 3.)	762-157
Control Wheel Neutral Tool (HF 622725)	N/A

MAIN GEAR AND DOORS

1. Troubleshooting

   See “Chart 1”.

2. Main Gear Oleo

   A. Disassembly

      The main gear oleo assembly may be removed from the gear oleo housing and disassembled with the gear removed from or installed in the airplane.

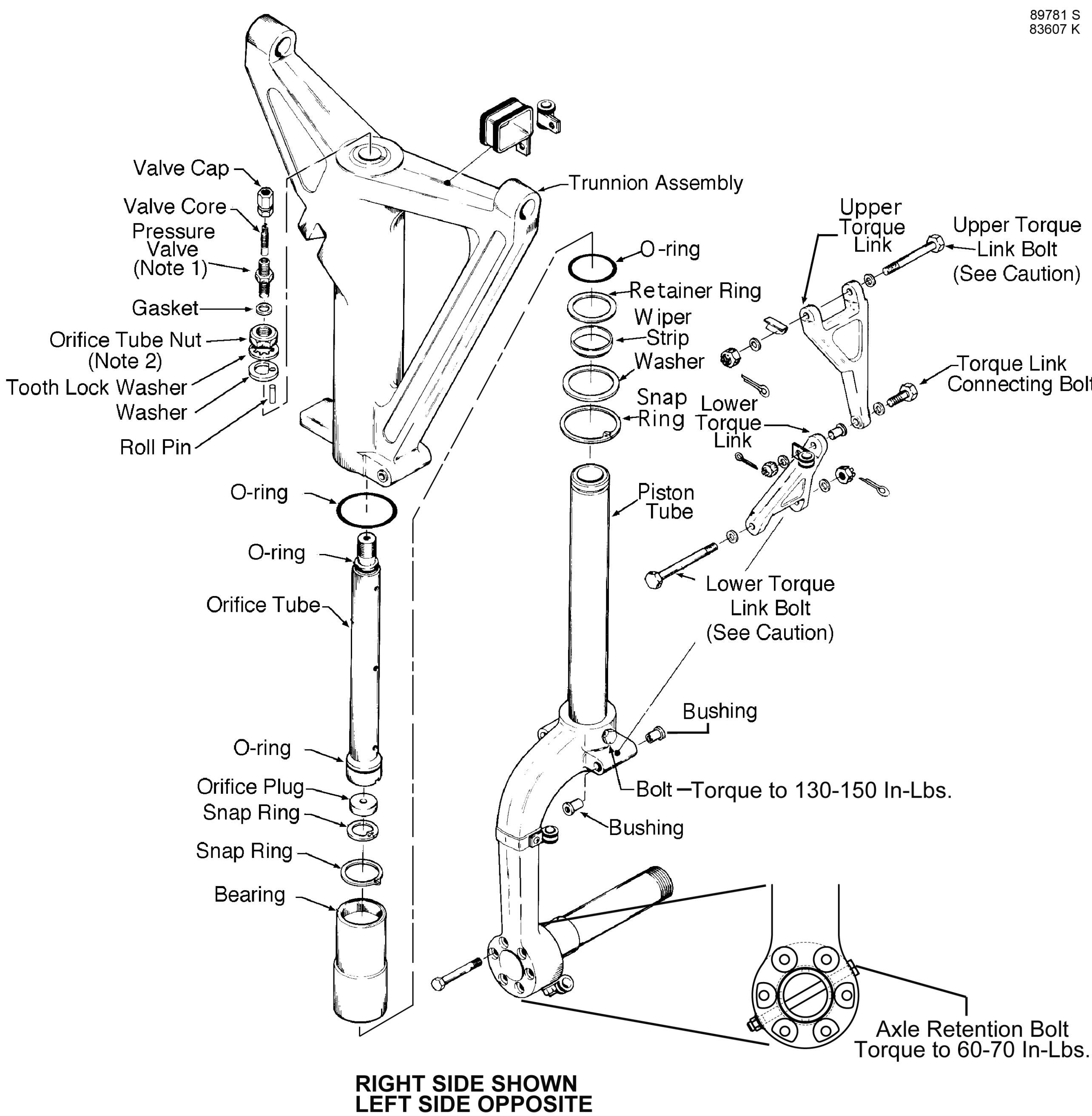
      (1) Place airplane on jacks (refer to 7-10-00).

      (2) Place a drip pan under the main gear to catch spillage.

      (3) Remove air and fluid from the oleo. Depress the air valve core pin to reduce strut pressure. Remove the filler plug. With a thin hose, siphon as much hydraulic fluid from the strut as possible.

CHART 1
TROUBLESHOOTING MAIN LANDING GEAR

Trouble	Cause	Remedy
Main landing gear shimmies during fast taxi, takeoff, or landing.	Tire out of balance.	Check balance and replace tire if necessary.
	Worn or loose wheel bearings.	Replace and/or adjust wheel bearings.
	Worn torque link bolts and/or bushings.	Replace bolts and/or bushings.
Excessive or uneven wear on main tires.	Incorrect operating pressure.	Inflate tire to correct pressure.
	Wheel out of alignment (toe in or out).	Check wheel alignment.
	Hydraulic actuator out of adjustment, allowing gear to slant in or out. (There is no adjustment on the extended length - adjust for retracted position only).	Check gear adjustment.
Strut bottoms on normal landing or taxiing on rough ground.	Insufficient air and/or fluid in strut.	Service strut with air and/or fluid.
	Defective internal parts in strut.	Replace defective parts.
Landing gear doors fail to completely close.	Landing gear not retracting completely.	Check adjustment of gear.
	Door retraction mechanism out of adjustment.	Check adjustment.



CAUTION: BE CERTAIN TO INSTALL BOLTS IN CORRECT DIRECTION TO PREVENT DAMAGE TO DOOR.

NOTE:
1. Apply Parker thread lube #6PB to threads of air valve upon installation.
2. Torque from 530 to 550 inch-pounds.
3. Apply Loctite 290 after pressing bushing on torque link.

Main Gear Oleo Strut Assembly
Figure 1

(4) Disconnect brake line at the joint located in the wheel well. Disconnect brake line retaining clamp that attaches the brake line to the upper part of the strut housing.

(5) To remove the piston tube assembly from the oleo housing, remove the cotter pin, nut, washers and bolt that connects the upper and lower torque links. Note the number and thickness of spacer washer(s) between the two links.

(6) Compress the piston tube. Reach up into the lower end of the oleo housing and release the snap ring from the annular slot at the bottom of the oleo housing.

(7) Pull the piston tube with its component parts from the cylinder housing.

(8) The piston tube components may be removed by removing the retainer ring from the top of the piston tube. Slide the bearing assembly (with its inner O-ring and ring), wiper strip and washer over the top of the piston tube.

(9) To remove the piston tube plug, located at the bottom of the piston tube assembly, remove the bolt and insert a rod into the hole in the bottom of the fork and push the plug out of the top of the piston tube.

(10) To remove the orifice tube, remove the orifice tube locknut, washer, lock washer and roll pin from the top of the orifice tube (at the top of the trunnion). Slide the orifice tube from the trunnion.

(11) The orifice tube plug may be removed by removing the snap ring from the slot at the bottom of the orifice tube.

B. Assembly

(1) Determine that all parts are cleaned and inspected.

(2) To install the piston tube plug, first lubricate the plug O-ring with hydraulic fluid and install it on the plug. Lubricate the inside wall of the piston tube. Insert the plug into the top of the piston tube and push it down the tube. Align the bolt holes of the fork, tube and plug. Then install the bolt assembly.

(3) If desired, cement a cork in the hole in the bottom of the fork body to prevent dirt from entering between the fork and tube.

(4) To assemble components of orifice tube, insert the orifice plate into the bottom of the tube and secure it with the snap ring. Install piston tube ring in slot on outer diameter of lower section of orifice tube.

(5) To install the orifice tube in the oleo housing, install upper O-ring on orifice tube and insert the tube up through the housing. With the end of the tube exposed through the top of the housing, install the washer, lock washer, roll pin and locknut. Tighten the locknut only finger tight at this time.

(6) Assemble components of piston tube on the tube by installing in order, snap ring, washer, wiper ring, bearing assembly (with outer O-ring and inner O-ring/ring) and retainer ring.

(7) Lubricate the wall of the cylinder oleo housing and piston tube. Carefully insert the piston tube assembly into the housing, guiding the orifice tube into the piston tube. Slide the washer and wiper strip into position and secure the assembly with the snap ring in the lower part of the trunnion.

(8) Torque the locknut at the top of the orifice tube per “Figure 1”.

(9) Ascertain that bushings are installed in the upper and lower torque link connection holes and then connect the upper and lower torque links by installing the bolt, washers, and nut. Be sure to use the same number of spacer washers noted during disassembly (to maintain wheel alignment). Tighten nut enough to permit no side play in the links, yet still allow bolt rotation. Secure nut with cotter pin.

(10) Connect brake line and bleed brakes.

(11) Lubricate gear assembly (refer to Lubrication Chart 3, 12-20-00).

(12) Compress and extend the strut several times to ascertain the strut will operate freely. The weight of the gear wheel and fork should allow the strut to extend.

(13) Service the oleo strut with fluid and air (refer to Oleo Struts, 12-10-00).

(14) Check main gear alignment (refer to Alignment of Main Landing Gear) and gear operation.

(15) Ascertain that gear is down and locked.

(16) Remove the airplane from the jacks.

C. Cleaning, Inspection, and Repair

(1) Clean all parts with a suitable dry-type cleaning solvent.

(2) Inspect landing gear oleo assembly components for the following:

(a) Bearings and bushings for excessive wear (refer to Main Gear Tolerance illustration for specifications), corrosion and overall damage.

(b) Snap rings for cracks, burrs, etc.

(c) Cylinder and orifice tube for corrosion, scratches, nicks and excessive wear.

(d) Orifice plate for hole restriction.

(e) Fork tube for corrosion, scratches, nicks, dents and misalignment.

(f) Air valve for general condition.

(3) Repair of the oleo is limited to smoothing out minor scratches, nicks or dents and replacement of parts.

3. Main Gear Assembly

A. Removal

(1) Place the airplane on jacks.

(2) Disconnect locking actuator from strut housing.

(3) Disconnect brake line at its upper end in the wheel well.

(4) Disconnect gear door actuating rod at the gear housing.

(5) Remove bolt which retains forward strut housing support to the forward trunnion pin.

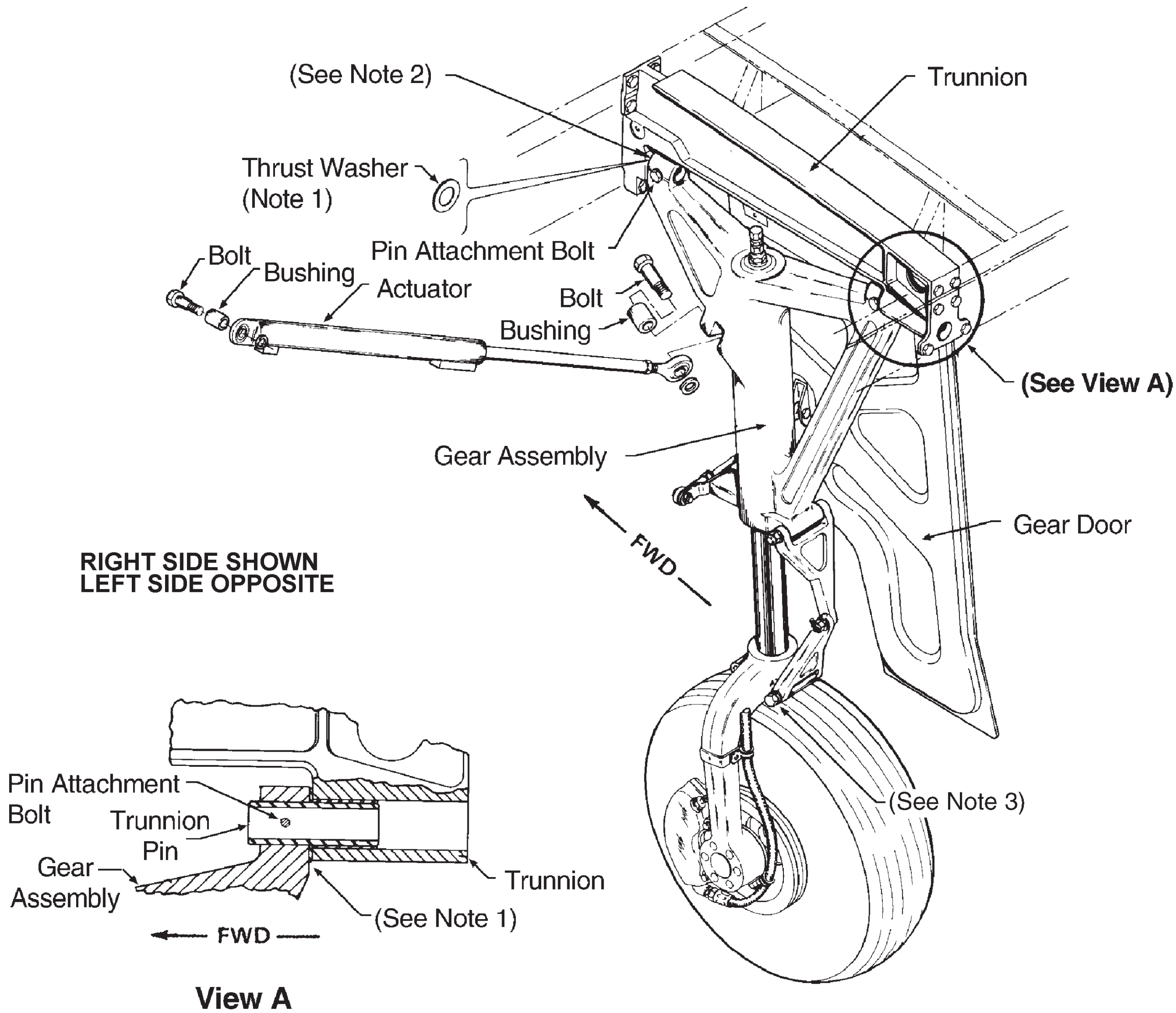
(6) Remove the forward trunnion pin by inserting a hook through the pin and drawing it towards the center of the gear assembly.

NOTE: When withdrawing the pin do not lose the teflon washer and note the amount of shims (if any) between the mating surfaces of the strut housing support and the support fitting.

(7) Remove the bolt which retains the aft strut housing support to the aft trunnion pin.

(8) Remove the aft trunnion pin by inserting a hook through the pin and drawing it towards the center of the gear assembly. Do not lose the teflon washer that was installed between the mating surfaces of the strut housing support and the support fitting.

(9) Allow the gear to slowly drop from the wing.



NOTE:
1. Install special thrust washers with the teflon surface towards the trunnion surfaces.
2. If required, shim between the forward gear attachment and rear surface of the thrust washer to obtain trunnion end play of .002 to .004 inches.
3. Add a maximum of one (1) each P/N 85012-133 and 85012-134 washers under head of lower torque link bolt to provide clearance between tail of bolt and main gear door during retraction, as required.

Main Gear Installation
Figure 2

B. Installation

NOTE: When assembling components of the landing gear, lubricate bearings, bushings and friction surfaces with proper lubricant as described in 12-10-00.

(1) Install gear housing in the wheel well of the wing as follows:

(a) Place the gear up into the wheel well and align the holes in the strut support with the holes in the support fittings.

NOTE: Ensure that one teflon washer is installed between each mating surface of the strut support and the support fittings. The teflon surface of the washer should face away from the centerline of the gear assembly.

NOTE: Install any previously removed shims between the mating surf aces of the forward strut support and the support fittings. There should be 0.002 - 0.004 of trunnion end play.

(b) Rotate the trunnion pin until the hole in the pin aligns with the hole in the strut support.

(c) Insert bolt (bolt head should be inboard) through strut support and trunnion pin. Secure bolt with nut and washers.

(d) Repeat steps (b) and (c) for remaining trunnion pin and strut support.

(2) Attach the gear door actuating rod to the gear housing.

(3) Connect brake line at its upper end in the wheel well.

(4) Connect locking actuator to gear housing.

(5) Ensure that the landing gear is lubricated per Lubrication Chart 3, 12-20-00.

(6) Check the alignment of the wheel per instructions in Alignment, below.

(7) Remove the airplane from the jacks.

C. Cleaning, Inspection, and Repair

(1) Clean all parts in a suitable dry-type cleaning solvent.

(2) Inspect the gear components for the following unfavorable conditions:

(a) Bolts, bearings and bushings for excessive wear, corrosion and damage.

(b) Gear housing, torque links and attachment plates for cracks, bends or misalignment.

(3) Check the general condition of each limit switch and its actuator. Check the electrical wiring for fraying, poor connections or conditions that may lead to failures.

(4) Repair of the landing gear is limited to reconditioning of parts such as replacing components, bearings and bushings, smoothing out minor nicks and scratches and repainting areas where paint has chipped or peeled.

D. Adjustment

(1) Place the airplane on jacks.

(2) Level the airplane laterally and longitudinally (refer to Leveling, 8-20-00).

(3) Disconnect gear door actuating rod from the strut housing by removing the rod attachment bolt. Secure the door out of the way.

(4) Disconnect the locking actuator from the strut housing.

(5) Adjust the locking actuator to a nominal length of 24.08 inches with no load on the wheel. Attach locking actuator to strut housing.

(6) Using the hydraulic pump, retract landing gear into wing well.

NOTE: This will require approximately 60 psi of pressure on the rod end part of the actuator to unlock the cylinder and allow the cylinder to retract.

(7) Adjust the locking cylinder rod end so that the fork is lightly contacting the rubber pad when the actuator is fully retracted.

(8) Inspect for adequate engagement of the rod end into the locking cylinder and tighten the check nut.

(9) If the down lock limit switch is properly adjusted, retract and extend the gear hydro-electrically to ascertain that the gear operates properly.

E. Alignment

See “Chart 2” on page 32108 and “Figure 3” on page 32108.

(1) Place a straightedge no less than twelve feet long across the front of both main landing gear wheels. Butt the straightedge against the tire at the hub level of the landing gear wheels. Devise a support to hold the straightedge in this position.

(2) Jack the airplane up just high enough to obtain a six and one-half inch dimension between the centerline of the strut piston and the centerline of the center pivot bolt of the gear torque links as shown in “Figure 3” on page 32108.

(3) Set a square against the straightedge and check to see if its outstanding leg bears on the front and rear side of the brake disc. (It may be necessary to remove the brake assembly to have clear access to the disc.) If it touches both forward and rear flanges, the landing gear is correctly aligned. The alignment for the main landing gear wheels is 0° 15’ ± 5’ toe-in.

NOTE: Measure alignment with struts compressed as described in Step (2) and with the wheel at the maximum toe-out extent of its freeplay.

NOTE: A carpenter’s square, because of its especially long legs, is recommended for checking main landing gear wheel alignment.

(4) If the square contacts the rear side of the disc, leaving a gap between it and the front flange, the wheel is toed-out. If a gap appears at the rear flange, the wheel is toed-in.

(5) To rectify the toe-in and toe-out condition, remove the bolt connecting the upper and lower torque links and remove or add spacer washers to move the wheel in the desired direction. See “Chart 2” on page 32108.

(6) Should a condition exist that all spacer washers have been removed and it is still necessary to move the wheel further in or out, then it will be necessary to turn the torque link assembly over. This will put the link connecting point on the opposite side allowing the use of spacers to go in the same direction.

(7) Recheck wheel alignment. If the alignment is correct, safety the castellated nut with cotter pin.

(8) If a new link on the top left main gear had to be installed or it had to be reversed during the alignment check, it will be necessary to check the gear safety switch (squat switch) bracket for engagement and locking in place.

(9) Check adjustment of landing gear safety switch (squat switch). (See 32-60-00, “Landing Gear Safety Switch” on page 32606.)

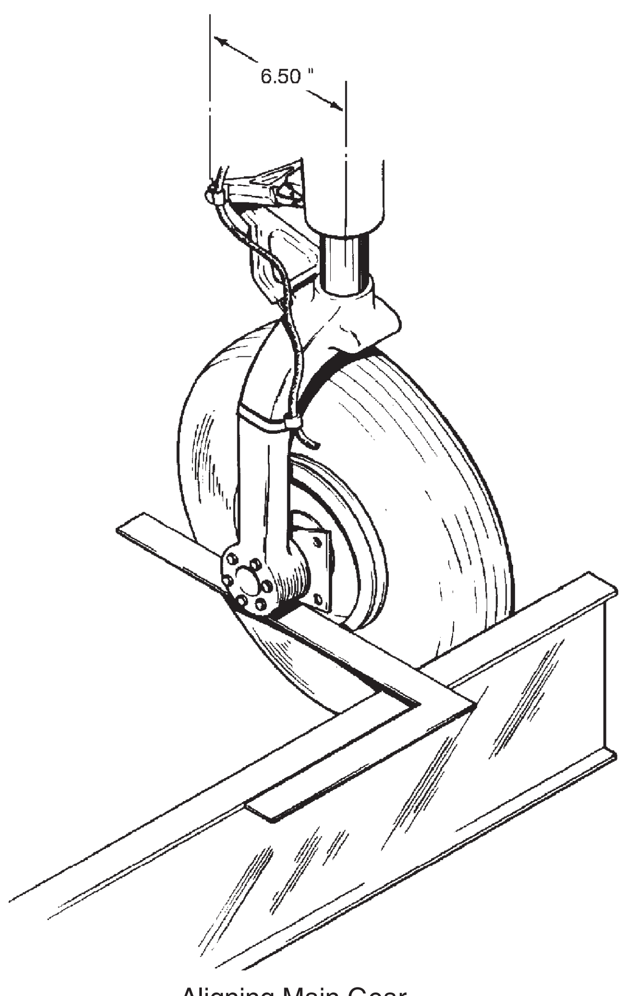
4. Main Gear Tolerances

See “Figure 4” on page 32109

CHART 2
TOE-IN / TOE-OUT CORRECTION CHART

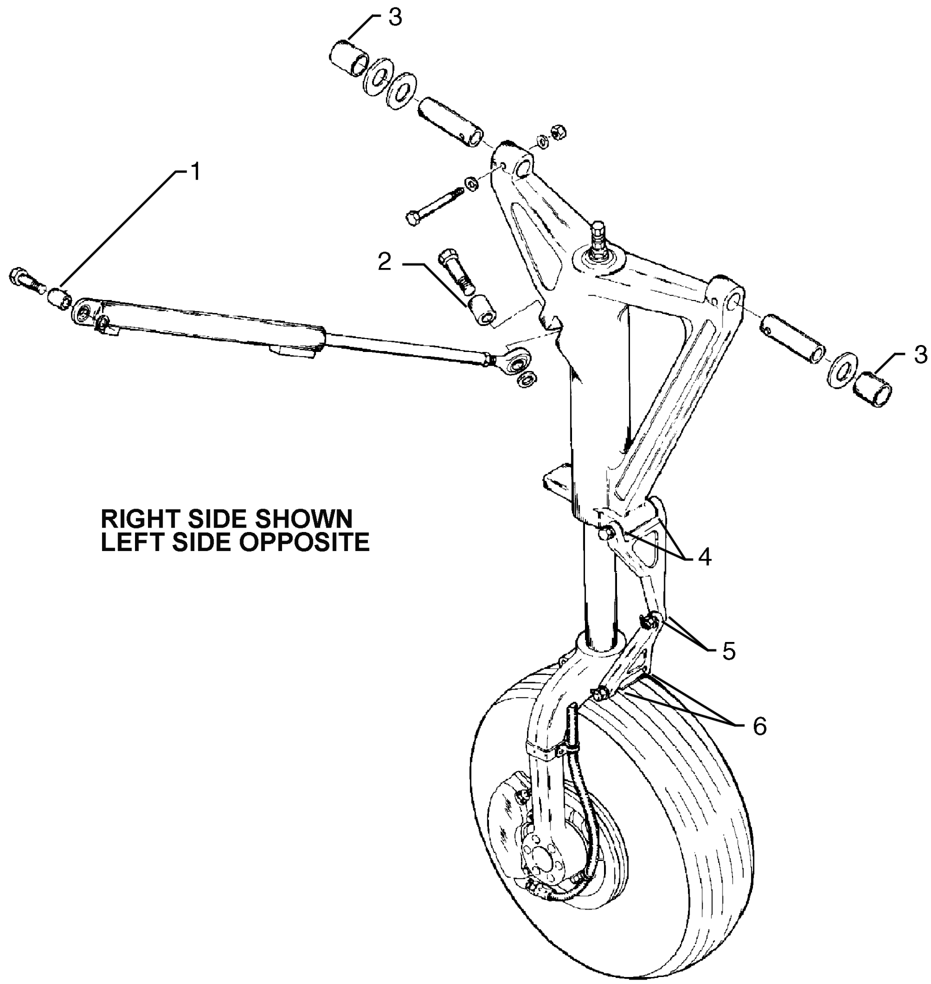
(PIR-89780T)

TOE-IN TOE-OUT ANGLE	SHIM WASHERS	WASHERS UNDER HEAD	WASHERS UNDER NUT	NAS6604 BOLT
0°		NAS1149F0463P	NAS1149F0463P (3)	D17
0° 15’	NAS1149F0432P	NAS1149F0463P	NAS1149F0463P (2)	D17
0° 33’	NAS1149F0463P	NAS1149F0463P	NAS1149F0463P (2)	D17
0° 48’	NAS1149F0432P NAS1149F0463P	NAS1149F0463P	NAS1149F0463P	D17
1° 04’	NAS1149F0463P (2)	NAS1149F0463P	NAS1149F0463P	D17
1° 19’	NAS1149F0432P NAS1149F0463P (2)	NAS1149F0432P	NAS1149F0463P	D17
1° 35’	NAS1149F0463P (3)	NAS1149F0463P	NAS1149F0463P (2)	D19
2° 05’ Max. Allow.	NAS1149F0463P (4)	NAS1149F0463P	NAS1149F0463P	D19



Aligning Main Gear
Figure 3

32108



Item No.	Part No.	Nomenclature	Manufacturer's Dimension (1)	Service Dimension (1)	Service Tolerance	Remarks
1	82695-59	Upper Actuator Attachment Bushing	.439 / .437			
3	16DU24	Trunnion Pin Bushing	1.00			
4	(FF411-4)	Lower Trunnion Bushing	.314 / .315			NOTES 2 & 3
5	31796	Torque Link Bushing	.252 / .251			NOTES 2 & 4
6	67026-7	Strut Bearing	.313 / .314			NOTES 2 & 3

Notes: 1. All dimensions are inside dimensions (ID), unless otherwise noted.

2. Line ream to this dimension after installation of part.

3. Press fit.

4. Wet install bushing with fluid resistant epoxy primer on adjacent surfaces of bushing and casting.

Main Gear Tolerances
Figure 4

5. Main Gear Door Assembly

A. Removal

(1) With the landing gear down, disconnect the door retraction rod from the door by removing the nut, washers and bolt.

(2) Remove the door from the wing panel by cutting the safety wire on the two hinge pins and driving the pins from the hinges.

B. Installation

(1) Position the door so that the hinges line up and insert the hinge pins. The pins should be installed with the safety wire holes towards the center line of the door.

(2) Ensure that the ends of the pins which have the safety wire holes protrude approximately 1/4 inch from the hinges. Safety the pins to the wing with MS20995-C41 safety wire.

(3) Attach the door retraction rod to the door with the previously removed bolt, washers and nut.

C. Cleaning, Inspection, and Repair

(1) Clean the door and retraction rod with a suitable cleaning solvent.

(2) Inspect the door for cracks or damage, loose or damaged hinges and brackets.

(3) Inspect the door retraction rod and end bearing for damage and corrosion.

NOSE GEAR AND DOORS

1. Troubleshooting

See “Chart 1”.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING NOSE LANDING GEAR

Trouble	Cause	Remedy
Nose landing gear shimmies during fast taxi, takeoff or landing.	Internal wear in bungee.	Replace bungee.
	Bungee loose at mounting.	Replace necessary parts and bolts.
	Tire out of balance.	Check balance and replace tire if necessary.
	Worn or loose wheel bearings.	Replace and/or adjust wheel bearings.
	Worn torque link bolts and/or bushings.	Replace bolts and/or bushings.
	Nose gear rake angle out of tolerance.	Adjust nose gear rake angle.
	Clearance between steering arm and nose gear steering rollers is out of tolerance.	Adjust clearance between steering arm and nose gear steering rollers.
	Incorrect tire pressure.	Inflate tire to correct pressure.
Excessive or uneven wear on nose tire.	Incorrect operating pressure.	Inflate tire to correct pressure.
	Wear resulting from shimmy.	Refer to preceding “trouble” for correction.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING NOSE GEAR

Trouble	Cause	Remedy
Nose gear fails to steer properly.	Oleo cylinder binding in strut housing.	Lubricate strut housing (refer to 12-10-00).
	Cylinder and/or strut housing bushings damaged.	Determine cause and correct.
	One brake dragging.	Determine cause and correct.
	Rollers not engaged. (Metal to metal in Gear Down Position.)	Determine cause and correct.
	Steering bellcrank loose on	Readjust and tighten. attachment plate.
	Steering bellcrank bearing and/or bolt worn.	Replace bearing and/or bolt.
	Bungee galling or binding.	Replace.
Nose gear fails to straighten when landing gear extends.	Steering arm roller sheared at top of strut.	Replace defective roller.
	Incorrect rigging of nose gear steering.	Check nose gear steering adjustment.
Nose gear fails to straighten when landing gear retracts.	Centering guide roller sheared.	Replace roller.
	Damaged guide.	Replace guide.
Strut bottoms on normal landing or taxiing on rough ground.	Insufficient air and/or fluid in strut.	Service strut with air and/or fluid.
	Defective internal parts in strut.	Replace defective parts.
Landing gear doors fail to completely close.	Landing gear not retracting completely.	Check adjustment of gear.
	Door retraction mechanism out of adjustment.	Check adjustment.

2. Nose Gear Oleo

A. Disassembly

The nose gear oleo assembly may be removed and disassembled from the gear oleo housing with the gear removed from or installed on the airplane.

(1) Place the airplane on jacks. (refer to Jacking, 7-10-00).

(2) Place a drip pan under the nose gear to catch spillage.

(3) Remove air and oil from the oleo strut. Depress the air valve core pin until the strut chamber pressure has diminished, remove the filler plug and, with a small hose, siphon as much hydraulic fluid from the strut as possible.

(4) To remove the complete strut and fork assembly from the oleo housing:

(a) Cut the safety wire on the three cap bolts that retain the steering arm to the strut. Remove the bolts.

(b) Remove the snap ring, thrust washer, and washers (if any) at the top of the strut housing.

(c) Pull the complete strut and fork assembly from the bottom of the housing.

(5) To remove the piston tube and fork from the strut:

(a) Separate the upper and lower torque links by removing the cotter pin, nut, washers, and bolt connecting the links. Separate the two links, noting the number of spacer washers between the two links.

(b) Compress the piston tube. Reach up along the tube and release the snap ring from the annular slot at the bottom of the strut.

(c) Pull the piston tube with its component parts from the strut.

(d) The piston tube components may be removed from the piston tube by removing the retainer from the annular slot at the top of the piston tube.

(e) Slide the bearing assembly (with inner O-ring and teflon retainer), wiper ring, washer and snap ring off of the piston tube.

(6) To remove the orifice tube:

(a) Remove the orifice tube retaining nut from the top of the strut assembly.

(b) Pull the orifice tube down and out of the strut.

(c) The orifice plate is removed from the bottom of the orifice tube by releasing the snap ring that holds the plate in position.

(7) To remove the piston tube plug with O-ring:

(a) Remove the bolt assembly.

(b) Insert a rod up through the hole in the body of the fork.

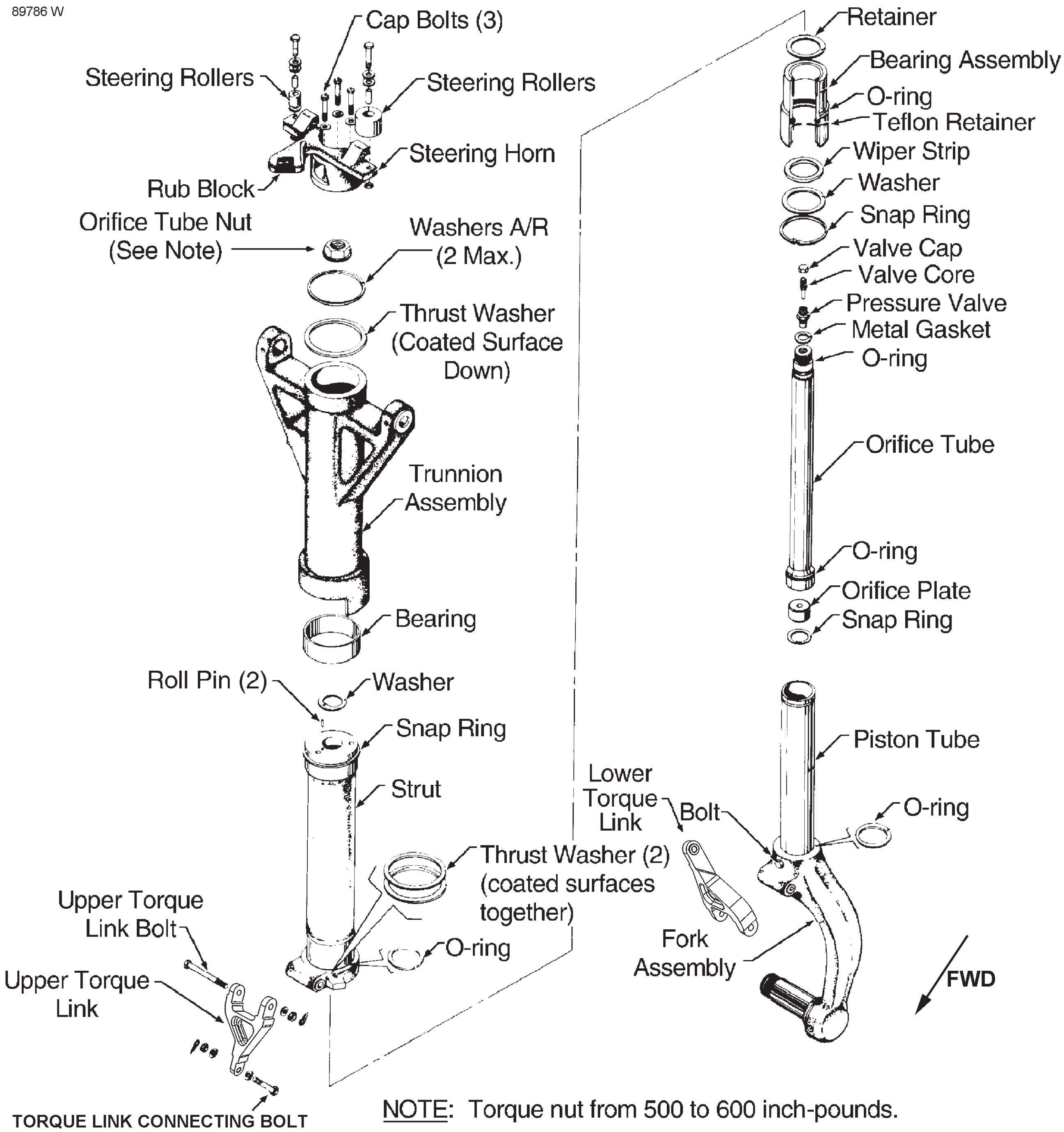
(c) Push the plug out through the top of the tube.

B. Assembly

(1) Ascertain that parts are cleaned and inspected.

(2) To install the piston tube plug, first lubricate the tube plug and O-ring with hydraulic fluid (MIL-PRF-5606) and install the O-ring on the plug. Lubricate the inside wall of the tube, insert the plug into the top of the tube and push it to the fork end. Align the bolt holes of the fork, tube and plug and install the bolt assembly.

(3) If desired, cement a cork in the hole in the bottom of the fork body to prevent dirt from entering between the fork and tube.



TORQUE LINK CONNECTING BOLT
(NAS6604D15 BOLT
MS14144-4 NUT
MS24665-132 COTTER PIN
NAS1149F0463P WASHER (2 REQ.))

NOTE: Torque nut from 500 to 600 inch-pounds.

Nose Gear Oleo Strut Assembly
Figure 1

(4) To assemble the components of the orifice tube:
   (a) Insert the orifice plate into the bottom of the tube, with the countersunk side of the hole in the orifice plate exposed.
   (b) Secure the plate with the snap ring, lubricate and install the two O-rings on the tube (one at each end).

(5) Insert the orifice tube up through the bottom of the strut. With the threaded portion of the orifice tube exposed through the top of the strut, install the washer and insert a roll pin through the washer into the strut. Install the orifice tube nut finger tight at this time.

(6) The fork and piston tube assembly may be assembled by installing the tube components on the tube. In order, slide onto the tube the snap ring, washer, wiper ring, bearing assembly (with inner O-ring and teflon retainer). Reinstall the retainer in the annular slot in the top of the piston tube.

(7) Lubricate the inner wall of the strut with hydraulic fluid. Insert O-ring into bottom of strut. Carefully insert the piston tube assembly into the bottom of the strut, allowing the orifice tube to guide itself into the piston tube. Push the tube all the way up in the cylinder. Position wiper ring and install wiper ring, slide washer up into the strut and secure the assembly by placing the snap ring into the annular slot in the bottom of the strut.

(8) At the top of the strut, torque the orifice tube retaining nut as specified in “Figure 1”.

(9) Ascertain that bushings are installed in the upper and lower torque link connection holes and then connect the upper and lower torque links by installing the bolt, washers, and nut. (See Figure 1 for proper hardware and bolt orientation.) Be sure to use the same number of spacer washers noted during disassembly. Tighten nut only tight enough to allow no side play in the links, yet still be free enough to rotate. Secure with cotter pin.

(10) Ensure that the two thrust washers (with their coated surfaces together) are installed in place on the lower portion of the strut immediately above the boss where the upper torque link attaches to the strut.

(11) Slide the strut assembly up into the trunnion. Ensure that a thrust washer (with its coated side down) is placed on top of the trunnion. Place the snap ring in its annular slot in the top of the strut.

(12) At the top of the oleo strut assembly install the steering arm to the top of the strut with the three cap bolts. Torque the bolts and safety them with 0.032 safety wire.

(13) Lubricate the gear assembly (refer to 12-20-00, Chart 4).

(14) Compress and extend the strut several times to ascertain that the strut will operate freely. Weight of the gear wheel and fork should allow the strut to extend.

(15) Service the oleo strut with fluid and air (refer to 12-10-00, Landing Gear Oleo Struts).

(16) Check nose gear for alignment (refer to Nose Gear Assembly - Alignment, below) and gear operation.

C. Cleaning, Inspection, and Repair

(1) Clean all parts with a suitable dry-type cleaning solvent.

(2) Inspect the landing gear oleo assembly components for the following:
   (a) Bearings and bushings for excess wear, corrosion, scratches and overall damage.
   (b) Lock rings for cracks, burrs, etc.
   (c) Cylinder and orifice tube for corrosion, scratches, nicks and excess wear.
   (d) Bushing assembly for looseness or turning in cylinder.
   (e) Orifice plate for hole restriction.
   (f) Fork tube for corrosion, scratches, nicks, dents and misalignment.
   (g) Air valve for general condition.

(3) Repair of the oleo is limited to smoothing out minor scratches, nicks and dents and replacement of parts.

3. Nose Gear Assembly

NOTE: This section incorporates the requirements from Piper Service Letter No. 1286C.

A. Removal

(1) Remove the engine cowling (refer to 71-10-00).

(2) Place the airplane on jacks (refer to 7-10-00).

(3) Retract the nose gear slightly to remove it from the down-locked position.

WARNING: USE HOSE CLAMPS TO HOLD DOWN SPRING WITHIN THE CONFINES OF SPECIAL TOOL. DO NOT ATTEMPT TO RELEASE DOWN SPRING WITHOUT THE USE OF THE SPECIAL TOOL ("FIGURE 2"). THE DOWN SPRING ASSEMBLY IS SPRING LOADED AND WILL FLY APART.

(4) Use the special tool ("Figure 2") to hold down spring at proper length for installation. Disconnect the actuator and emergency down spring mechanism from the right-hand side of the trunnion by removing the bolt, spacers and washers.

(5) The landing gear may then be removed by removing the nuts, washers and bolts that attach the gear trunnion to the attachment points on each side of the engine mount.

B. Installation

NOTE: When assembling any units of the landing gear, lubricate bearings, bushings, and friction surfaces with the proper lubricant as described in 12-20-00.

(1) To install the landing gear assembly, position the gear so that the bolt attachment points on the housing align with the attachment points on the engine mount. Install pivot bolts, washers and nuts. Tighten the nuts to a snug fit, still allowing the gear to swing free.

(2) Connect the actuator and emergency down mechanism to the right-hand side of the trunnion with the previously removed bolt, spacers and washers. Safety the bolt to the trunnion with MS20995-C41 safety wire. Remove the down spring retaining tool.

(3) Cycle the gear a minimum of three times to ensure proper operation.

(4) Ascertain that gear is lubricated per 12-20-00, Chart 4.

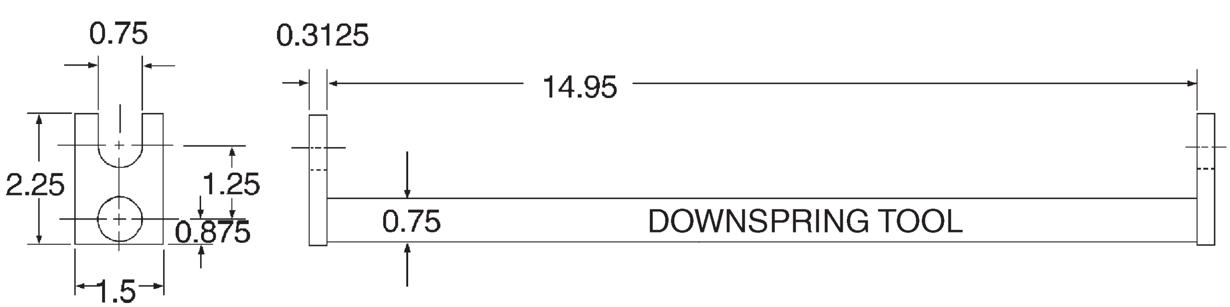
(5) Check the adjustment of the gear per Nose Gear Assembly - Adjustment, below.

(6) Install engine cowling (refer to 71-10-00).

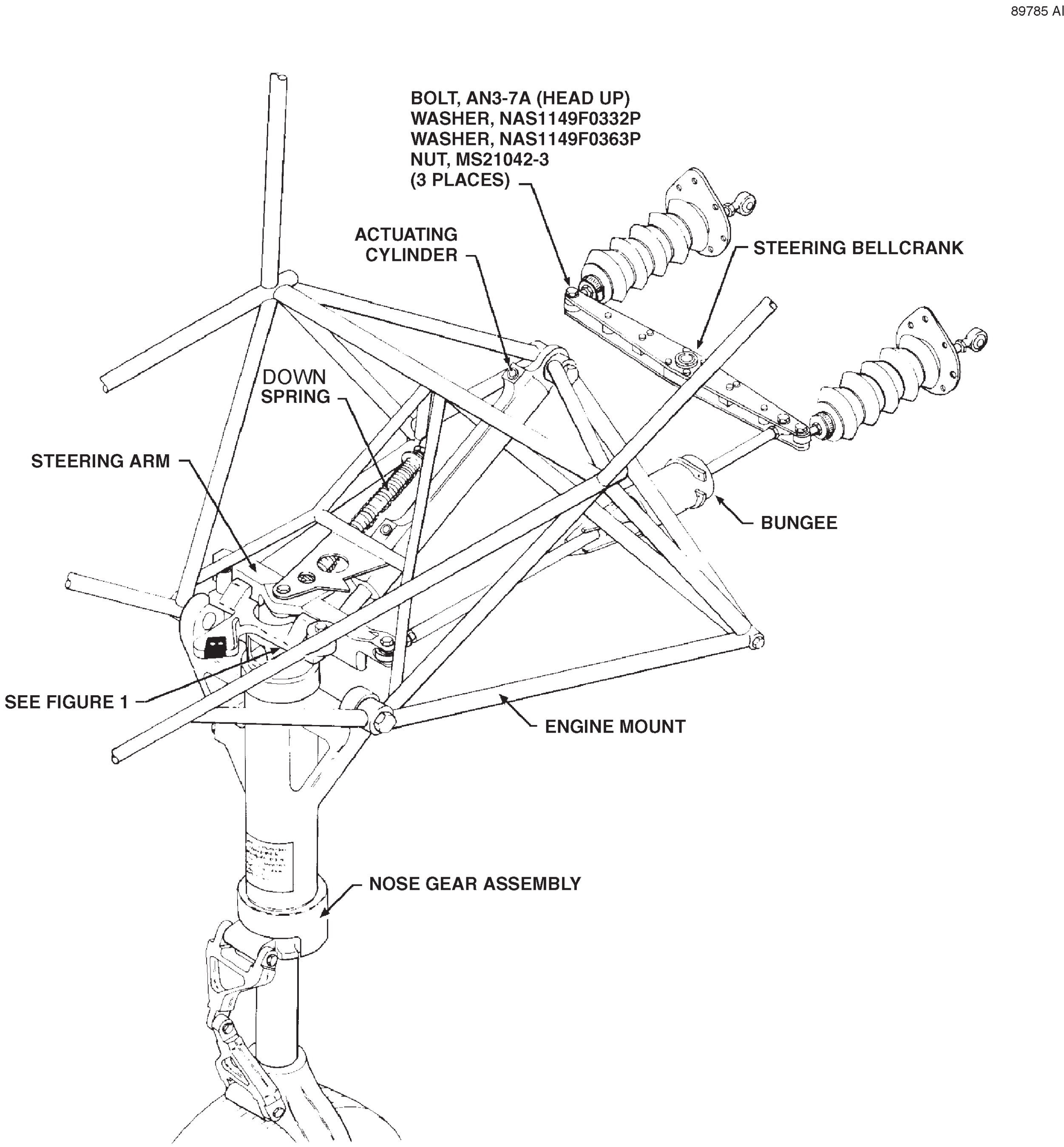
(7) Retract landing gear and check door operation per Nose Gear Door Adjustments, below.

(8) Check the alignment of the nose gear per "Alignment" on page 32208.

(9) Remove the airplane from the jacks (refer to 7-10-00).



Nose Gear Downspring Remove/Install Tool (P/N 762-157)
Figure 2



Nose Gear Installation
Figure 3

C. Cleaning, Inspection, and Repair

(1) Clean all parts with a suitable dry-type cleaning solvent.

(2) Inspect the gear components for the following unfavorable conditions:

(a) Bolts, bearings and bushings for excessive wear, corrosion and damage.

(b) Gear housing, torque links and down spring for cracks, bends or misalignment.

(3) Check the general condition of each limit switch and its actuator. Check wiring for fraying, poor connections or conditions that may lead to failures.

(4) The bungee requires no service other than routine inspection. In case of damage or malfunction, the bungee should be replaced rather than repaired.

(5) Repair to the landing gear is limited to reconditioning of parts such as replacing bearings and bushings, smoothing out minor nicks and scratches, repainting of areas where paint has chipped or peeled and replacement of parts.

D. Alignment

NOTE: In addition to maintaining the nose landing gear alignment, maintaining the main landing gear alignment affects proper ground directional control. Refer to 32-10-00 for the main landing gear alignment procedure.

(1) Place the airplane on a smooth level floor that will accommodate the striking of a chalk line.

(2) Place the airplane on jacks (refer to 7-10-00).

(3) Level the airplane laterally and longitudinally (refer to 8-20-00).

(4) From the center point of the tail skid, extend a plumb bob and mark the contact point on the floor.

(5) Extend a chalk line from the mark on the floor below the tail skid to a point approximately three feet forward of the nose wheel. Allow the line to pass under the wheel at the centerline of the tire. Snap the chalk line.

WARNING: THE FOLLOWING STEP RESTRICTS THE MOVEMENT OF RUDDER PEDALS WITH USE OF A DOWEL ROD. THE DOWEL ROD MUST BE REMOVED FROM THE RUDDER PEDALS ONCE ALIGNMENT OF THE NOSE GEAR ASSEMBLY IS COMPLETED. FAILURE TO REMOVE THE DOWEL ROD WILL RESULT IN LOSS OF CONTROL IN THE RUDDER AND NOSE LANDING GEAR.

(6) Align the rudder pedals in the lateral position with a dowel rod. (Refer to “Figure 5” on page 322013.) Depress the left pedal and insert the dowel rod into the right pedal tube as far as it will go. Release the left pedal and draw out the dowel rod enough to insert it into the left pedal’s tube until it stops. Set the rudder pedals in their neutral position per “Figure 5” and verify that pedal alignment is maintained.

(7) Check to ensure that the steel bellcrank mounted in the nose gear tunnel is perpendicular (crosswise) to the sides of the tunnel (steering pushrods from the bellcrank to the pedals must be equal length).

(8) Disconnect steering bungee at the forward end.

(9) To align the nose wheel straight forward, stand in front of the nose gear and align the center rib of the tire with the chalk line, or lay a straightedge along the side of the tire and parallel the straightedge with the chalk line.

(10) Adjust the steering bungee length and install with nose wheel straight forward. Check that the rod ends have sufficient thread engagement by ascertaining that a wire will not go through the check hole in the rod. Tighten the jam nuts.

(11) To check the nose gear steering for its 30° right and left maximum travel, mark the 30° angles on each side of the nose wheel from the centerline and wheel pivot point. Turn the wheel to its maximum travel in both directions to check for allowable travel. Should travel be too much in one

direction but not enough in the other direction, check for possible damage to the gear fork or torque links.

WARNING: THE DOWEL ROD MUST BE REMOVED FROM THE RUDDER PEDALS ONCE ALIGNMENT OF THE NOSE GEAR ASSEMBLY IS COMPLETED. FAILURE TO REMOVE THE DOWEL ROD WILL RESULT IN LOSS OF CONTROL IN THE RUDDER AND NOSE LANDING GEAR.

(12) Remove the dowel rod from the aligned rudder pedals previously installed in step "(6)" on page 32208.

(13) As part of the alignment procedure, it is necessary to check the rake angle and, if required, adjust it. See "Adjustment" on page 32209.

E. Adjustment

(PPS50056-5, Rev. J.)

NOTE: It is necessary to inspect and/or adjust the nose gear rake angle as part of the nose gear alignment procedure. In addition, inspection and/or adjustment of the rake angle is necessary whenever the pilot experiences poor steering, or upon the replacement or service to the nose gear trunnion, the nose gear actuator, or the engine mount.

(1) Place the airplane on jacks (refer to Jacking 7-10-00).

(2) Place gear in DN position. Extended position is determined by the cylinder locking (refer to 32-30-00, Nose Gear Actuator Assembly).

(3) Verify that the nose gear rake angle is set between zero (vertical) and one-half degree forward, as shown in "Figure 4" on page 322011. Using a prop protractor, digital level, or other suitable tool, measure the relative angle between the nose gear piston tube and the lower (belly) skin of the fuselage, just aft of the nose gear wheel well. When properly rigged, this angle will measure between 90.0 and 90.5 degrees. If necessary, adjust the length of the fully extended actuator to achieve the required nose gear rake angle as follows:

(a) Loosen the jam nut(s) that secure the rod end bearing to the forward end of the actuator.

(b) Adjust the overall length of the rod end/actuator assembly to achieve the required nose gear rake angle.

(c) Tighten jam nut(s).

NOTE: If unable to rig nose gear rake angle with two jam nuts installed, remove one jam nut. Maximum thread exposure with single nut is 0.28 inch.

(4) Remove airplane from jacks.

(5) Any adjustment to the nose gear rake angle will also affect the clearance between the steering arm and the nose gear steering rollers. Verify/adjust this clearance as described in "Clearance", below.

F. Clearance

Adjust the steering arm to achieve the desirable clearance between it and the steering horn rollers. First, adjust it with the assembly in neutral position, then with the left pedal pressed and, finally, with the right pedal pressed. See "Figure 6" on page 322013 and "Figure 7" on page 322014.

(1) Clearance with the Rudder Pedals in the Neutral Position

(a) With the landing gear in the down and locked position, weight proportionally on the nose gear and the nose wheel facing forward, adjust the steering arm.

(b) Loosen the steering arm bolt assembly and adjust the jam nuts fore and aft (refer to View B–B of "Figure 4" on page 322011).

(c) Place two (2) 0.020 in. thick feeler gauges between the steering horn rollers and the steering arm. Adjust the steering arm to achieve contact between the steering horn roller, the feeler gauge, and steering arm, for both left and right hand sides simultaneously.

(d) Ensure that there is a minimum of 0.010 in. gap between both steering horn pads and the steering arm. When properly adjusted, the acceptable clearance between both rollers and the steering arm is between 0.010 and 0.030 in., as shown in “Figure 6” on page 322013.

NOTE: This clearance requirement applies to both rollers at the same time. The clearance measurement is to be taken while the nose wheel is locked in alignment with the longitudinal axis, and the steering arm is parallel to the steering horn pads (as shown in View A–A of “Figure 4” on page 322011).

(e) Torque the steering arm assembly nut per torque note in View B–B of “Figure 4” on page 322011, and tighten the locking assembly jam nuts.

NOTE: In Steps (2) and (3), below: the small roller or right hand steering horn pad will make contact with the steering arm when the left rudder pedal is depressed; the larger roller or left hand steering horn pad will make contact with the steering arm when right rudder pedal is depressed.

(2) Clearance with the Left Rudder Pedal Depressed:

(f) With the nose wheel off the ground and free to turn, and the left rudder pedal pressed firmly against the left forward stop, determine if there is: a 0.010 in. minimum clearance between the steering arm and the larger roller, and a positive clearance between the steering arm and the left hand steering horn pad, on the steering horn assembly, as shown in “Figure 7” on page 322014.

(g) If the clearance is less than the minimum 0.010 in., then complete Steps (c) and (d), below. Increase the nose landing gear centered clearance per the steering arm adjustment procedure, Step (1), above. A 0.010 in. minimum clearance must be maintained for the larger roller on the steering horn assembly at full left rudder pedal deflection.

(h) With weight on wheels, lock the rudder pedals together, in alignment with each other and on fuselage station 105.70. See “Figure 5” on page 322012.

(i) To align the nose wheel straight forward (weight remaining on wheels), stand in front of the nose gear and align the center rib of the tire with the chalk line. Alternatively, lay a straightedge along the side of the tire and parallel the straightedge with the chalk line.

NOTE: The chalk line referred to in step (d) is the same chalk line created in “Alignment” on page 322010.

(j) If Steps (c) and (d) were necessary, unlock the rudder pedals, then repeat Step (a), above, to confirm that minimum clearance is achieved.

(3) Clearance with the Right Rudder Pedal Depressed:

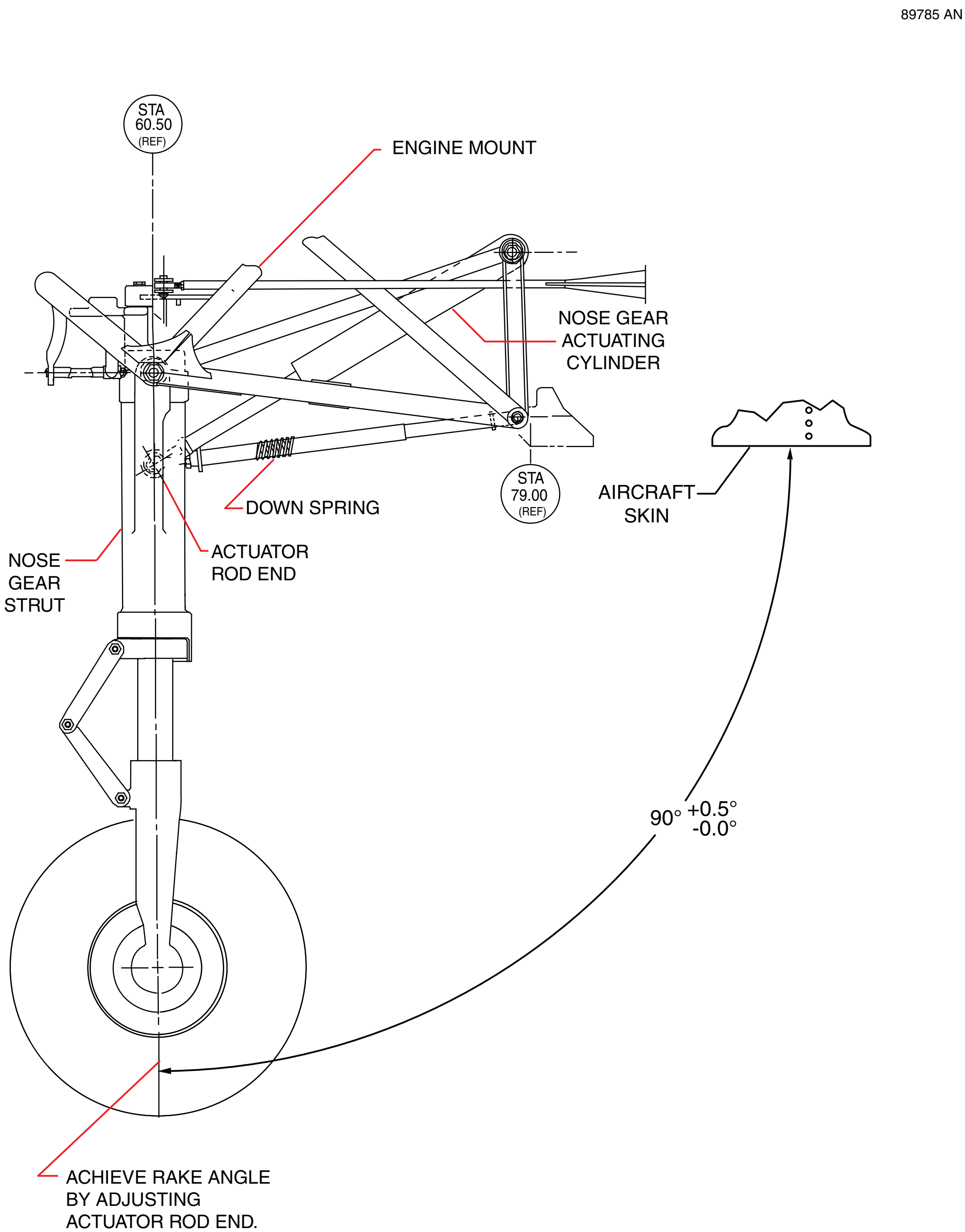
(a) With the nose wheel off the ground and free to turn, and the right rudder pedal pressed firmly against the right forward stop, determine if there is: a 0.010 in. minimum clearance between the steering arm and the small roller, and a positive clearance between the steering arm and the right hand steering horn pad, on the steering horn assembly, as shown in “Figure 7” on page 322014.

(b) If the clearance is less than the minimum 0.010 in., then complete Steps c) and d) under Step 2), above. Increase the nose landing gear centered clearance per the steering arm adjustment procedure, Step (1), above. A 0.010 in. minimum clearance must be maintained for the small roller, and a positive clearance for the right hand steering horn pad, on the steering horn assembly, at full right rudder pedal deflection.

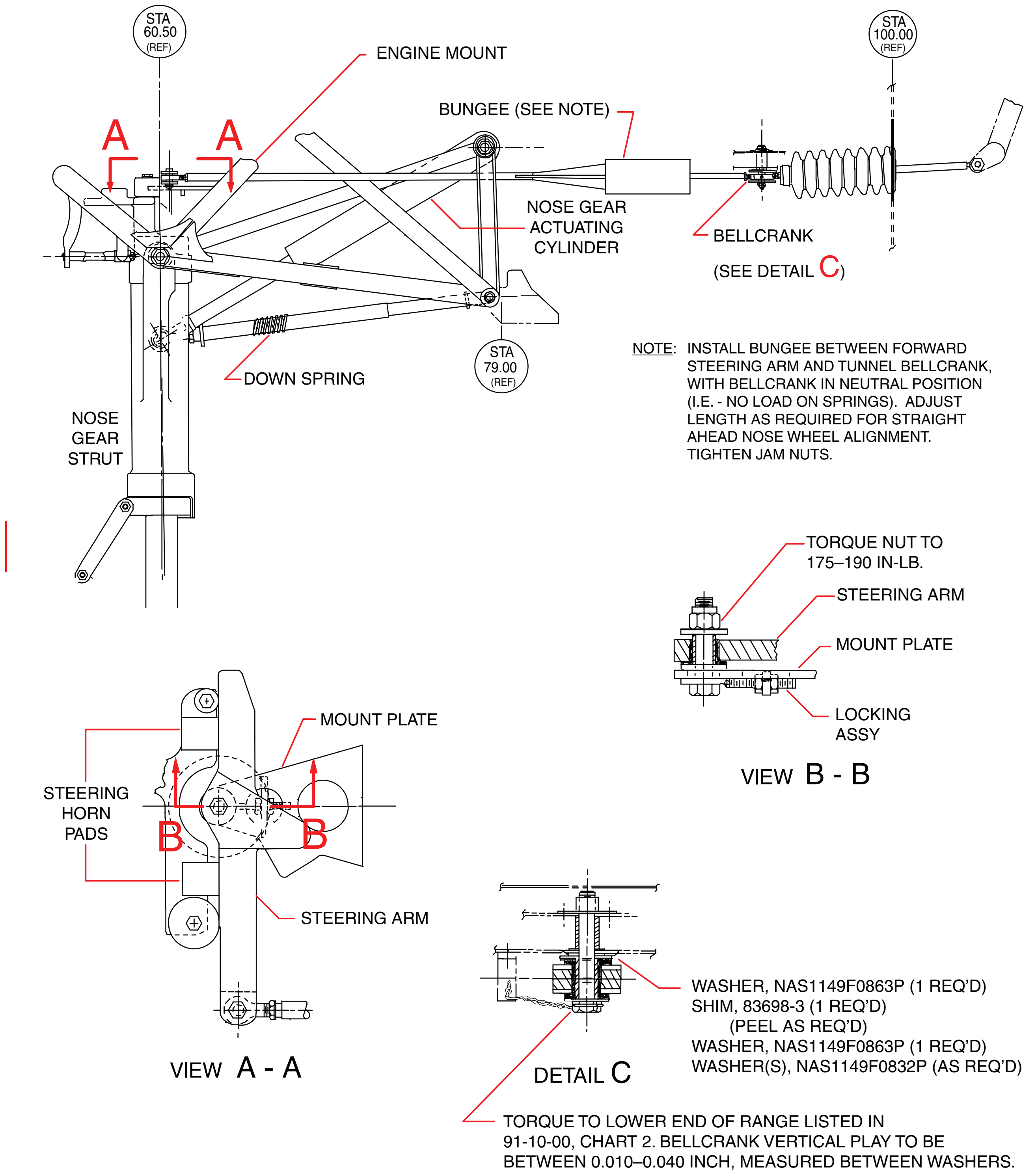
(c) If Steps (c) and (d) under Step (2) were necessary, unlock the rudder pedals, then repeat Step a), above, to confirm that minimum clearance is achieved.

G. Rudder Cable Tension

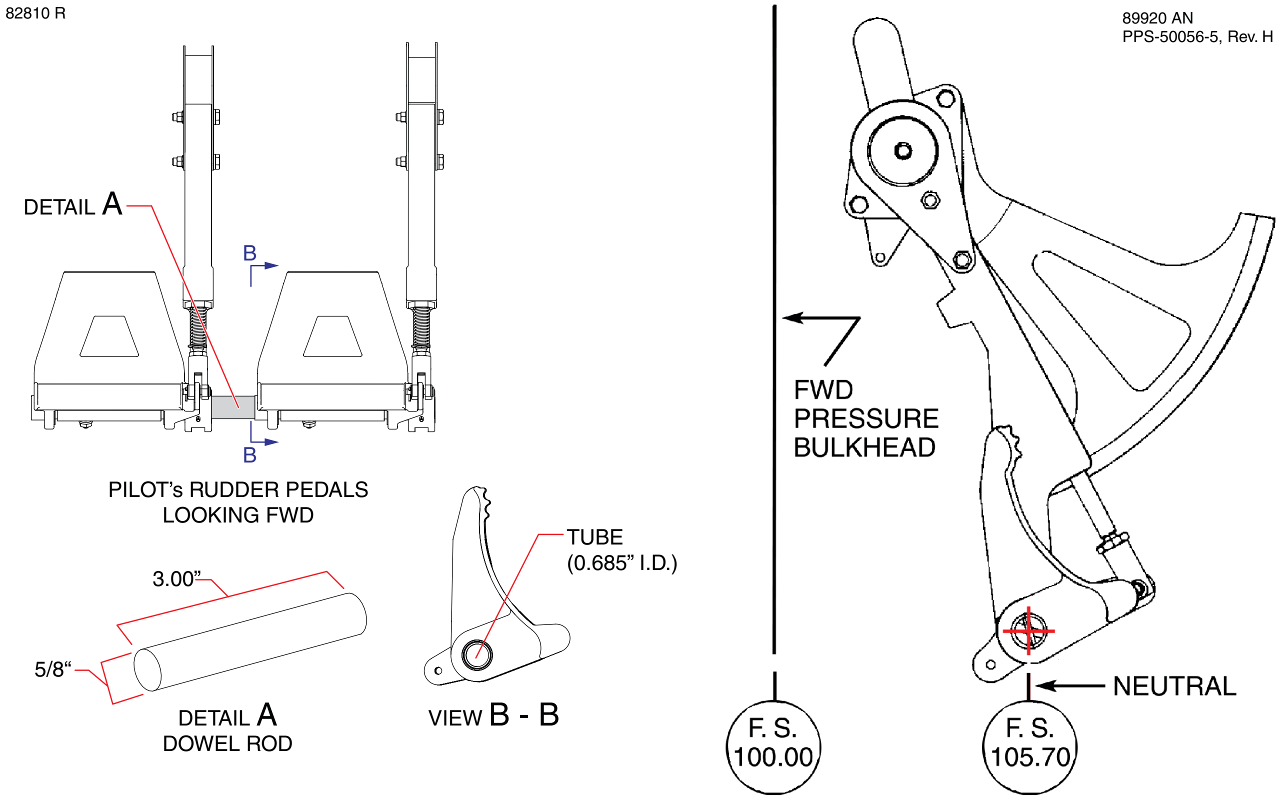
Confirm that the tension of both rudder control cables is correct, per the appropriate steps in 27-20-00.



Nose Gear Adjustment
Figure 4 (Sheet 1 of 2)



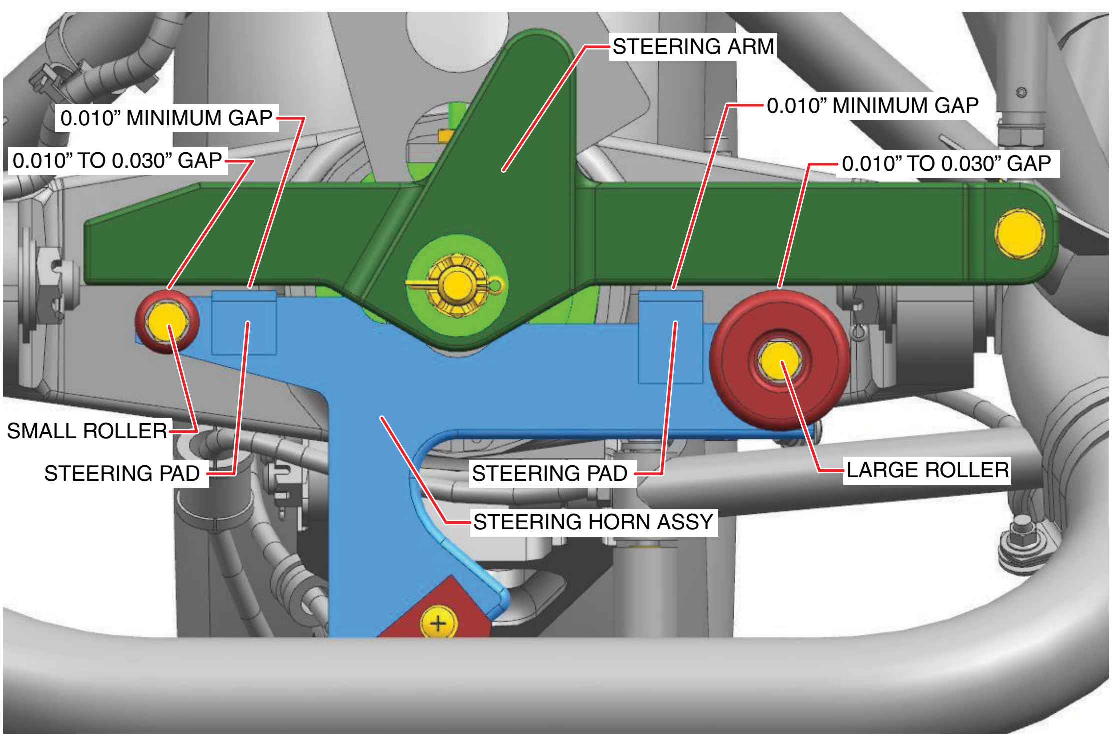
Nose Gear Adjustment
Figure 4 (Sheet 2 of 2)



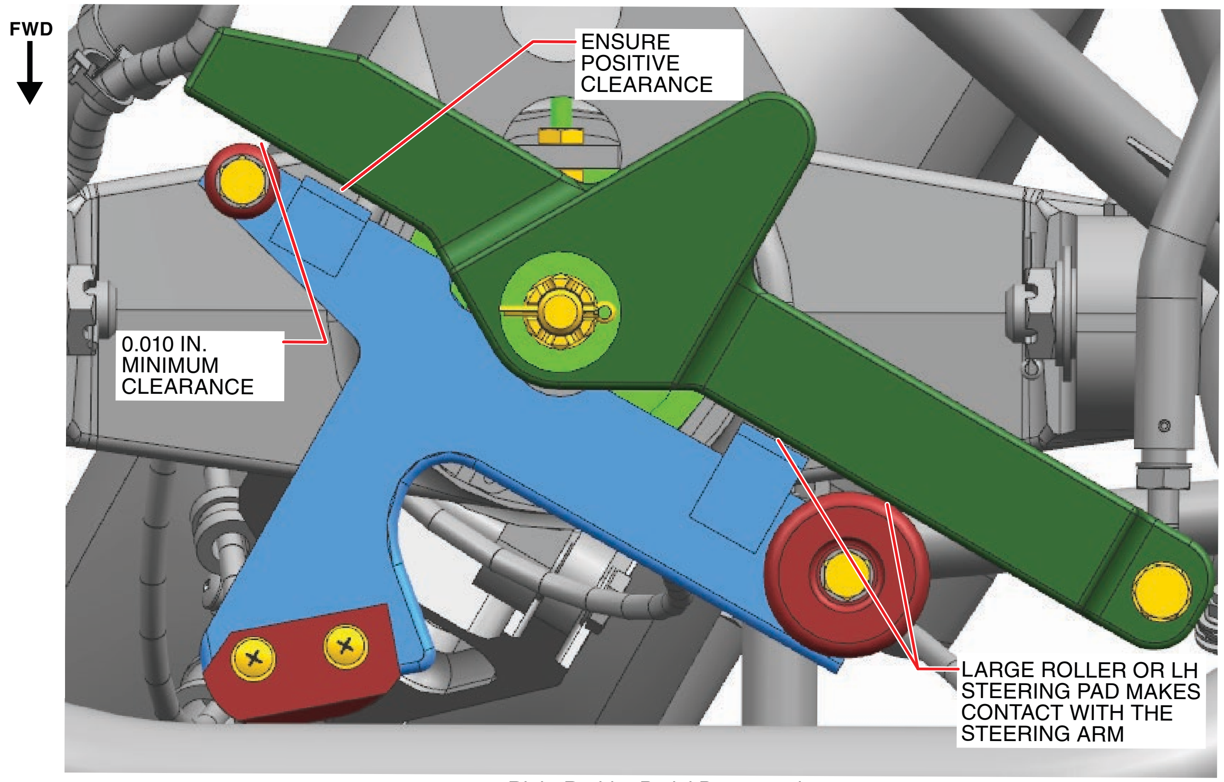
Rudder Pedals Aligned

Rudder Pedals at Neutral Position

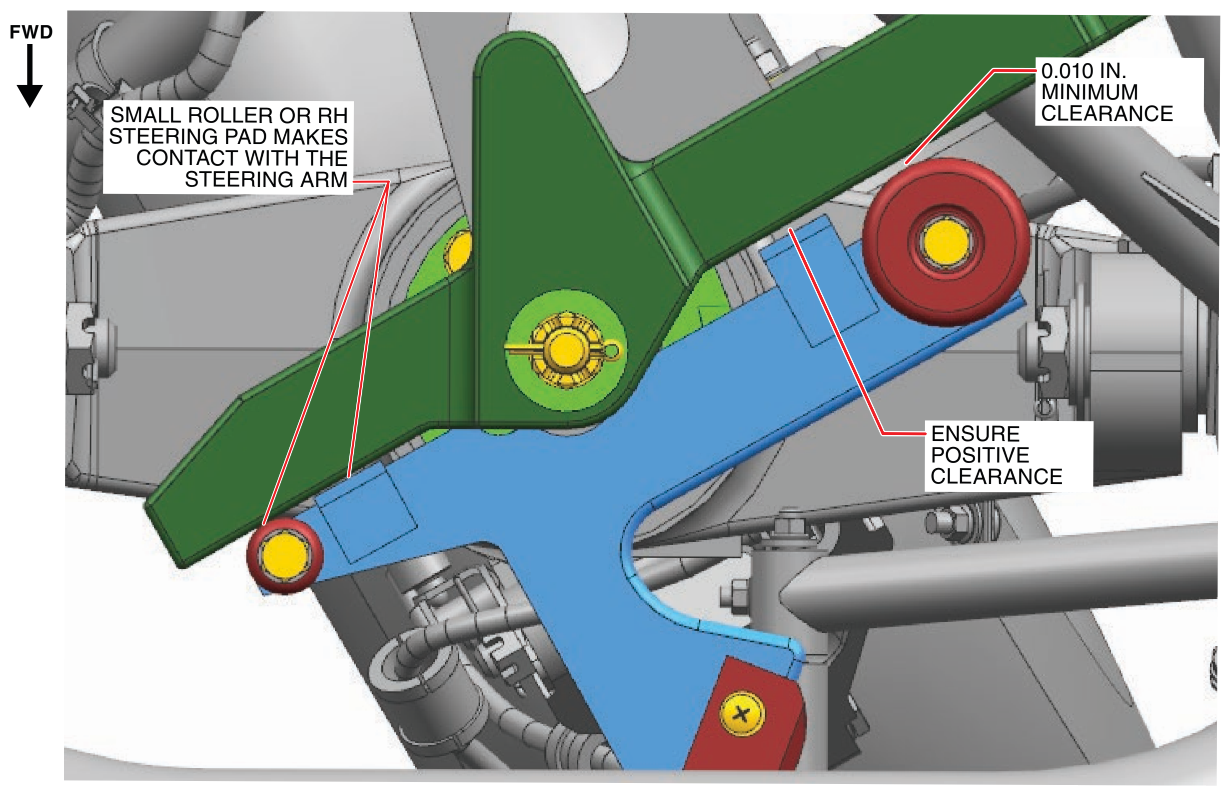
Rudder Pedals in Neutral Position
Figure 5



Steering Mechanism in Neutral Position (Looking Down)
Figure 6



Right Rudder Pedal Depressed



Left Rudder Pedal Depressed

Steering Mechanism in Depressed Positions (Looking Down)
Figure 7

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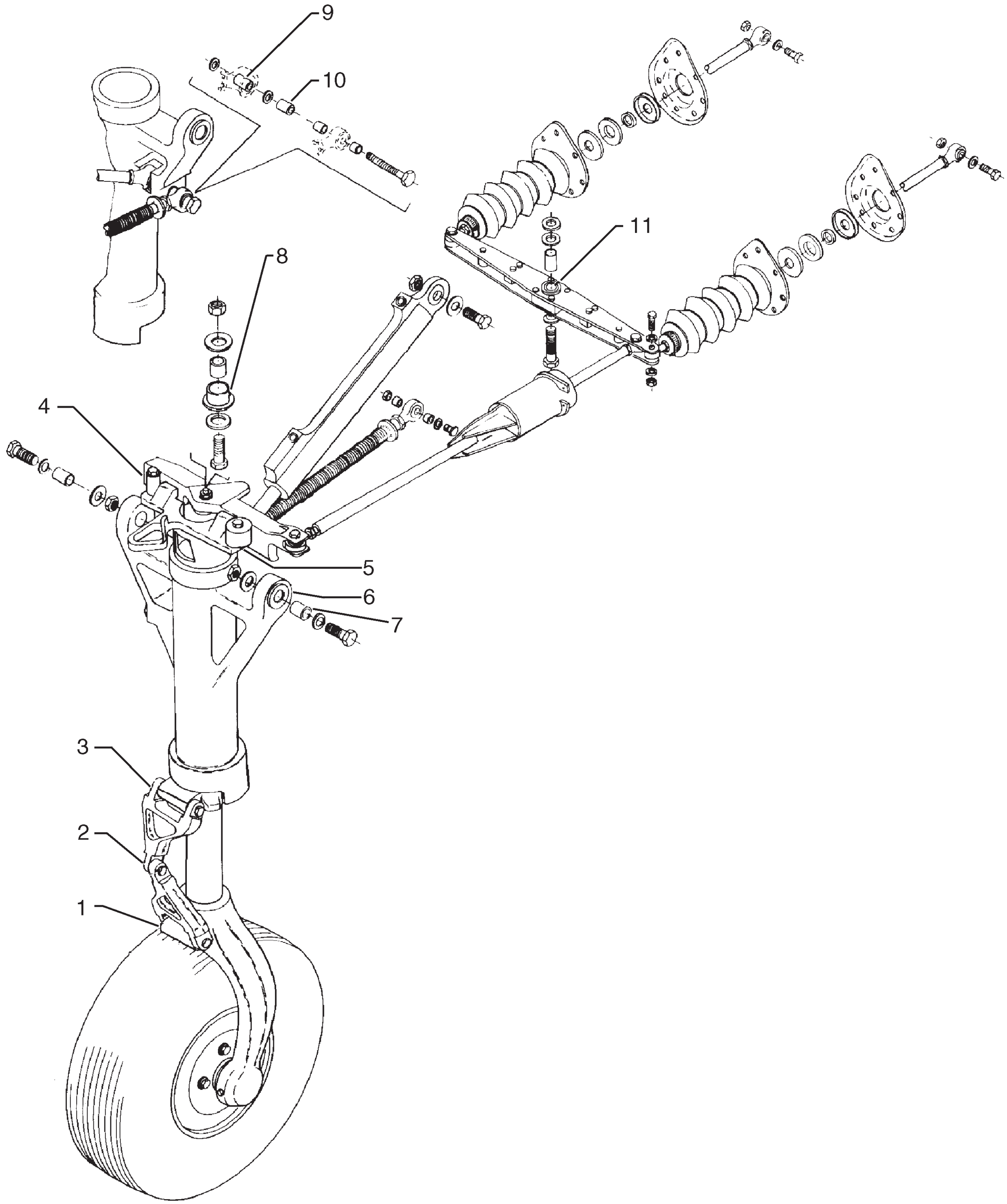
4. Nose Gear Tolerances

See “Figure 8”.



Nose Gear Tolerances
Figure 8 (Sheet 1 of 2)

Item No.	Part No.	Nomenclature	Manufacturer's Dimension (1)	Service Dimension (1)	Service Tolerance	Remarks
1	452-857 (FF411-2)	Lower Strut Bearing	.312 / .313			NOTES 2 & 4
2	452-450 (FF310-5)	Link Assembly Bearing	.2495 .2505			NOTES 2, 3 & 4
3	452-857 (FF411-2)	Upper Strut Bearing	.314 / .313			NOTES 2 & 5
4	63900-35	R/H Roller Bushing	.312 O.D.			
5	86700-47	L/H Roller Bushing	.312 O.D.			
6	83630-12	Trunnion Support Bushing	.753			NOTE 4
7	82695-3	Trunnion Spacer	.501 / .500			
8	83630-4	Steering Arm Bearing	.501 / .502			NOTE 2
9	82695-61	Actuator Spacer Bushing	.377 / .375			
10	82695-62/108	Emergency Down Mechanism Spacer Bushing	.377 / .375			
11	83630-2	Steering Bellcrank Pivot Bearing	.502 / .501			NOTE 2

NOTES: 1. All dimensions are inside dimensions (ID), unless otherwise noted.
2. Line ream to this dimension after installation of part.
3. Install with Loctite 290.
4. Shrink fit.
5. Press fit.

Nose Gear Tolerances
Figure 7 (Sheet 2 of 2)

5. Nose Gear Door Assembly

See “Figure 9” on page 322019.

A. Removal

(1) With nose gear extended, disconnect door link by removing nut, bolt and washers.

(2) To remove doors from cowl, bend one end of both hinge pins straight and pull out both pins.

B. Installation

(1) Install the gear doors by aligning the hinge halves of the door and the door support assembly and then inserting the two hinge pins. It is recommended that new pins be used. Bend the ends of the pins to secure them in place. Rotate the hinge pin ends so that they point into the wheel well when the doors are closed.

(2) Attach the door link to the door with the previously removed bolt, nut and washers.

C. Cleaning, Inspection, and Repair

(1) Clean all parts with a suitable cleaning solvent.

(2) Inspect doors for damage, loose or damaged hinges and brackets.

(3) Inspect door retraction link assemblies and arms for damage or wear.

(4) Check the tension spring (right door) for wear and tension. Reject the spring if the tension does not maintain the doors in the full open position.

(5) Repairs to the door are limited to replacement of hinges and painting.

(6) Repairs to the retraction mechanism are limited to replacement of parts, removal of light corrosion and painting.

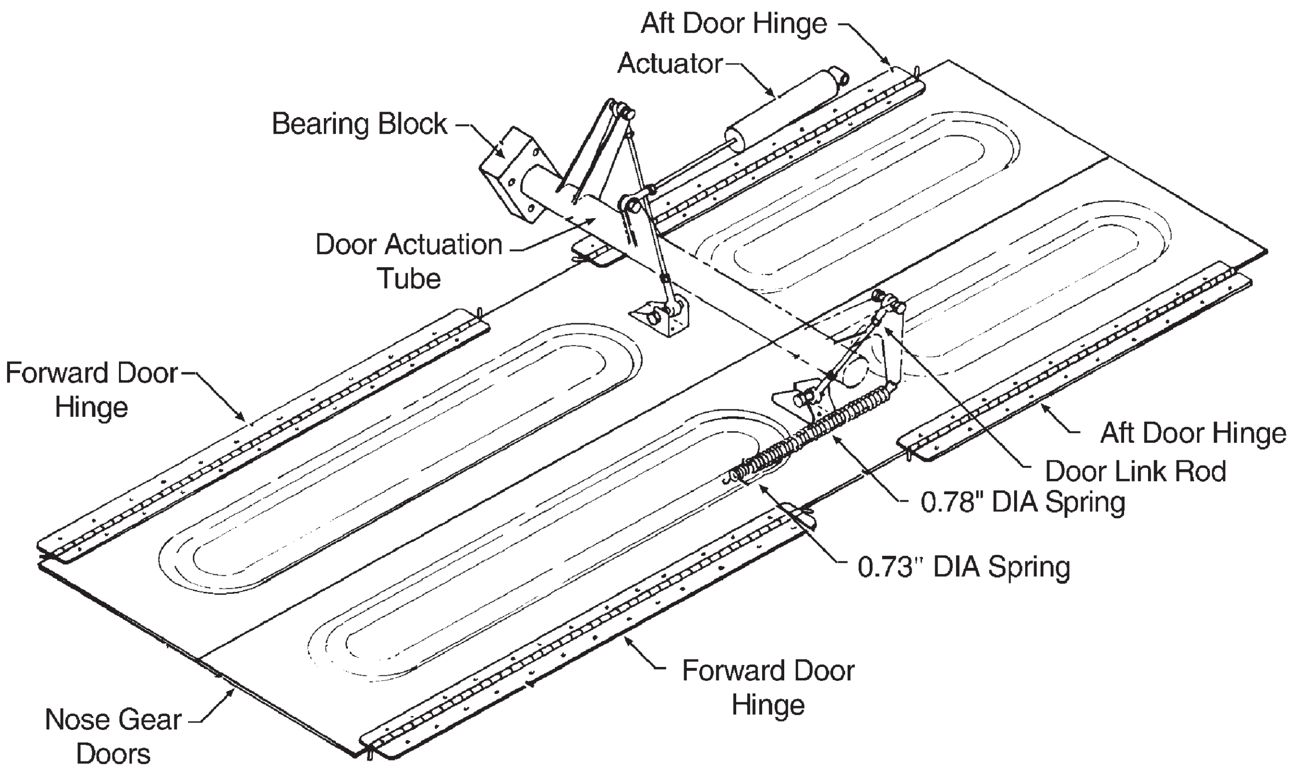
6. Nose Gear Door Adjustments

A. Place the airplane on jacks (refer to 7-10-00).

B. Disconnect both gear door actuating rods.

C. Retract gear into wheel well.

D. Adjust rod assemblies so that the door assemblies form a straight line at B.L. 0.000. Adjust the right hand door first by itself, the adjust the left hand door.



Nose Gear Doors
Figure 9

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EXTENSION AND RETRACTION

1. Troubleshooting

   See “Chart 1”.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING LANDING GEAR EXTENSION AND RETRACTION SYSTEMS

Trouble	Cause	Remedy
Landing gear system fails to operate.	Hydraulic pump circuit breaker open.	Reset circuit breaker and determine cause.
	Hydraulic pump circuit wires broken.	Check wiring.
	Safety (squat) switch out of adjustment.	Adjust switch.
	Squat switch inoperative.	Replace switch.
	Pressure switch inoperative.	Check bypass circuitry.
		Replace switch.
	Hydraulic pump inoperative.	Check ground wire.
		Repair or replace pump.
	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid to full level and investigate cause.
	Battery low or dead.	Check condition of battery. Recharge or replace.
	Hydraulic fluid leak in system.	Isolate leak and repair.
	Hydraulic pump electric motor inoperative.	Repair / replace motor top plate assembly.
	Hydraulic lines obstructed or leaking.	Check lines.
Landing gear retraction extremely slow.	Hydraulic pump motor.	Check pump motor.
	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid. Check for leaks.
	Internal / external leak in system.	Isolate and repair / replace.
	Restriction in hydraulic lines.	Isolate and check hydraulic lines.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING LANDING GEAR EXTENSION AND RETRACTION SYSTEMS

Trouble	Cause	Remedy
Pump stops during gear retraction / extension.	Hydraulic pump circuit breaker opens.	Reset circuit breaker and determine cause for overload.
	Pressure switch out of adjustment.	Check bypass circuitry.
		Remove and readjust or replace switch.
	Bypass circuitry malfunction or nose gear extension prior to main gear extension.	Check per Landing Gear Retraction System Functional Test, below.
	Pressure relief valve out of adjustment.	Adjust / replace pressure relief valve.
	Mechanical restriction or obstruction in hydraulic system to allow pressure to build up and shut off pump before gear has extended / retracted.	Place airplane on jacks and run extension check. Isolate and determine cause.
Pump fails to shut off though gear has fully retracted / extended.	Pressure switch inoperative.	Repair / replace switch.
	Pressure switch out of adjustment.	Check bypass circuitry.
		Adjust switch.
	Pump relay sticking.	Replace relay.
	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid. Check for leaks.
	Pressure relief valve out of adjustment.	Adjust / replace pressure relief valve.
	Internal / external leakage of system.	Check gear actuating cylinders for internal / external leakage.
		Check for internal damage to hydraulic pump.
		Check for damaged hydraulic lines.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING LANDING GEAR EXTENSION AND RETRACTION SYSTEMS

Trouble	Cause	Remedy
Pump running intermittently after gear has retracted/ extended.	Pump leaking.	Isolate and repair.
	Internal leakage of system.	Check free fall valve for internal leakage.
		Check pressure relief valve.
		Check gear actuating cylinders for internal leakage.
	External leakage of system.	Check for leak between reservoir and pump.
		Check gear actuating cylinders for external leakage.
		Check for broken or damaged hydraulic lines.
Pump running intermittently after gear has retracted.	Hydraulic fluid low in reservoir.	Fill reservoir and check for leaks.
Gear stops part way up, but pump continues to run.	Hydraulic fluid in reservoir below operating level.	Fill reservoir with hydraulic fluid.
	Internal leakage of system.	Check gear actuating cylinders for internal leakage.
		Check free fall valve for internal leakage.
		Check for broken or damaged hydraulic lines.
	Emergency extender pulled.	Reset emergency extender.
All gears fail to free fall.	Free fall function of selector fails to operate.	Check mechanical connections.
		Inspect valve for damage.
	Gear pivot points dry.	Lube pivot points.

2. Landing Gear Retraction System Functional Test

WARNING: BE SURE AIRPLANE IS ON JACKS BEFORE PROCEEDING WITH THE FOLLOWING TEST.

The following tests verify that the landing gear retraction system is functioning properly.

A. S/N's 4636001 thru 4636651 less 4636633; S/N's 4692001 and Up

(PIR-PPS60033-7, Rev. G/-9, Rev. P/-13, Rev. F.)

(1) Setup

(a) Verify that the main and nose gears are properly adjusted (see 32-10-00 and 32-20-00, respectively).

(b) Place the airplane on jacks per 7-10-00.

(c) Connect a 28 Vdc regulated power source capable of supplying a minimum of 50 amperes to the aircraft electrical system.

NOTE: With power supplied through the auxiliary power plug the batt-master switch should remain in the "OFF" position. The APU circuit bypasses the batt-master switch to energize the ship power circuits.

NOTE: Before starting test, check for adequate clearance of nose gear lights, sequence valves, actuators, hoses, etc.

NOTE: Never operate the emergency gear release piston with the hydraulic pump circuit breaker "IN".

(d) Set the sequence valve.

(e) Adjust the plunger screw outward until the sequence lever depresses the plunger approximately .125 inch when the nose gear is fully retracted, and the nose gear door actuating torque tube rotates. Lock the plunger and screw in place and connect the nose gear door actuation rods. These rods shall be long initially and shortened progressively until the nose gear doors seat firmly, but are not distorted when closed. Cycle the gear as required to ensure proper operation.

(f) During all tests, add clean filtered MIL-PRF-5606 (Revision "F" or higher) hydraulic fluid to the reservoir as necessary to keep fluid level just above the "L" mark on the sight gauge / see-thru reservoir oil level decal with the landing gear in the down and locked position. Verify that the fluid level does not exceed the "F" mark on the reservoir when the landing gear is in the up and locked position.

(g) Check/adjust switches.

1) Check/adjust squat switch per Landing Gear Safety Switch (Squat Switch) Adjustment, 32-60-00.

2) Check to ensure that the gear door closed (gear "UP") switches are wired to the proper panel indicator light (gear warning annunciator).

(h) Bleed the hydraulic system per 29-10-00, Servicing Hydraulic System.

(2) Procedure

CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE HYDRAULIC PUMP IS OPERATING. ALWAYS PULL THE HYDRAULIC PUMP CIRCUIT BREAKER BEFORE TESTING THE EMERGENCY GEAR RELEASE SYSTEM. FAILURE TO COMPLY WILL DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

CAUTION: IN S/N'S 4636314 AND UP AND S/N'S 4692001 AND UP, DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE LANDING GEAR ARE DOWN AND LOCKED. DOING SO CAN DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

CAUTION: DO NOT OVERFILL AND DO NOT ALLOW HYDRAULIC FLUID LEVEL TO DROP BELOW THE "L" (LOW) MARK ON THE SIGHT GAUGE / SEE-THRU RESERVOIR OIL LEVEL DECAL WITH THE GEAR IN THE DOWN POSITION.

NOTE: Perform the following in the sequence given. Any failure of the system to respond as specified indicates a malfunction which must be corrected before proceeding further.

(a) In S/N's 4636001 thru 4636374: Pull the throttle aft to the closed position.
In S/N's 4636375 and up and S/N's 4692001 and up: Jump G8L20 to G8K20 at the manifold pressure switch connector.

(b) Place the gear selector switch in the DOWN position.

(c) Verify that:

1) Three green safe lights are illuminated.
2) Red gear warning light is off.
3) Gear warning horn does not sound.
4) Hydraulic pump does not operate.
5) I. E. - Leave the gear down for at least five (5) minutes. Check that the hydraulic pump motor does not operate. If the pump operates at any time during the five (5) minute period, check for leaks or malfunctioning components in the system.

NOTE: One momentary pump operation is allowable during this five (5) minute period, provided that the gear warning light is out and no repeated pump operations for a subsequent fifteen (15) minute period occur.

(d) Place the gear selector switch in the UP position.

(e) Verify that:

1) Three green safe lights are out.
2) Red gear warning light illuminates.
3) In S/N's 4636375 and up and S/N's 4692001 and up: The hydraulic pump light illuminates while gear are in transit. (Not applicable to G1000 aircraft).
4) Gear warning horn sounds.
5) All gear retract fully and the gear doors close.
6) Hydraulic pump motor stops operating.
7) In S/N's 4636375 and up and S/N's 4692001 and up: The hydraulic pump light goes off. (Not applicable to G1000 aircraft).

(f) In S/N's 4636001 thru 4636374: Advance the throttle to the mid-travel position. The warning horn should stop sounding and the gear warning light should go out.

In S/N's 4636375 and up and S/N's 4692001 and up: Remove jumper between G8L20 to G8K20 at the manifold pressure switch connector. The warning horn stops sounding.

(g) With the gear fully retracted extend the flaps to the 10° position. The red “GEAR WARN” annunciator and gear warning horn should not operate. At flap extension beyond 10° to the full down position the red “GEAR WARN” annunciator and the gear warning horn should operate simultaneously and continuously.

Operate the flap extension back to the 10° position and the red “GEAR WARN” annunciator will turn off and gear warning horn will stop sounding.

(h) Leave the gear up for at least five (5) minutes. Check that the hydraulic pump motor does not operate. If the pump operates at any time during the five (5) minute period, check for leaks or malfunctioning components in the system.

NOTE: One momentary pump operation is allowable during this five (5) minute period, provided that the gear warning light is out and no repeated pump operations for a subsequent fifteen (15) minute period occur.

CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE HYDRAULIC PUMP IS OPERATING. ALWAYS PULL THE HYDRAULIC PUMP CIRCUIT BREAKER BEFORE TESTING THE EMERGENCY GEAR RELEASE SYSTEM. FAILURE TO COMPLY WILL DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

(i) Pull the “Hydraulic Pump” circuit breaker OUT.

(j) Release the guard and pull the emergency gear release (i.e. - free fall) knob OUT.

(k) Verify that:

1) The landing gear falls to the down and locked position.
2) Red gear warning light goes out when the gear locks.
3) Three green safe lights are illuminated when the gear locks.
4) In S/N’s 4636314 and up (i.e. - airplanes equipped with Frisby hydraulic pumps only): allow pressure to bleed off before continuing (approximately two (2) minutes).

NOTE: The main gear may not fall completely into the locked position during ground free fall tests. Extend the main gear using a side force at the center line of axle that does not exceed nine (9) pounds. The gear should move easily and then lock in place.

(l) Place the gear selector switch in the DOWN position.

(m) Push in the emergency gear release (i.e. - free fall) knob and engage the guard.

(n) Push the “Hydraulic Pump” circuit breaker IN.

(o) Check that three green lights stay on.

NOTE: Pump may run momentarily.

(p) Cycle gear up and down, one time to verify normal operations.

(q) Place the landing light switch ON.

(r) Place the gear selector switch in the UP position.

(s) Verify that:

1) Hydraulic pump motor operates and the gear retracts.
2) Three green gear safe lights go out.
3) Red gear warning light illuminates, until all the gear are up, and then goes out.

4) The gear retracts fully in:
   a) In S/N's 4636001 thru 4636313 (i.e. - airplanes equipped with Parker Hannifin hydraulic pumps only) - eight (8) seconds or less.
   b) In S/N's 4636314 and up and S/N's 4692001 and up (i.e. - airplanes equipped with Frisby/Triumph hydraulic pumps only) - nine (9) seconds or less.
5) Hydraulic pump motor stops operating after the gear is up.
6) Gear warning horn does not sound.
7) The landing light shuts off as the gear door closes.
8) Place the gear selector switch in the DOWN position, verify three (3) green safe lights are illuminated and red gear warning light goes out.

(t) Gear Down Pressure Switch Bypass Relay Circuit Test:
1) In S/N's 4636001 thru 4636313 (i.e. - airplanes equipped with Parker Hannifin hydraulic pumps only):
   a) Disconnect the gear down pressure switch circuit and conduct the Gear Selector Switch DOWN Check, below.

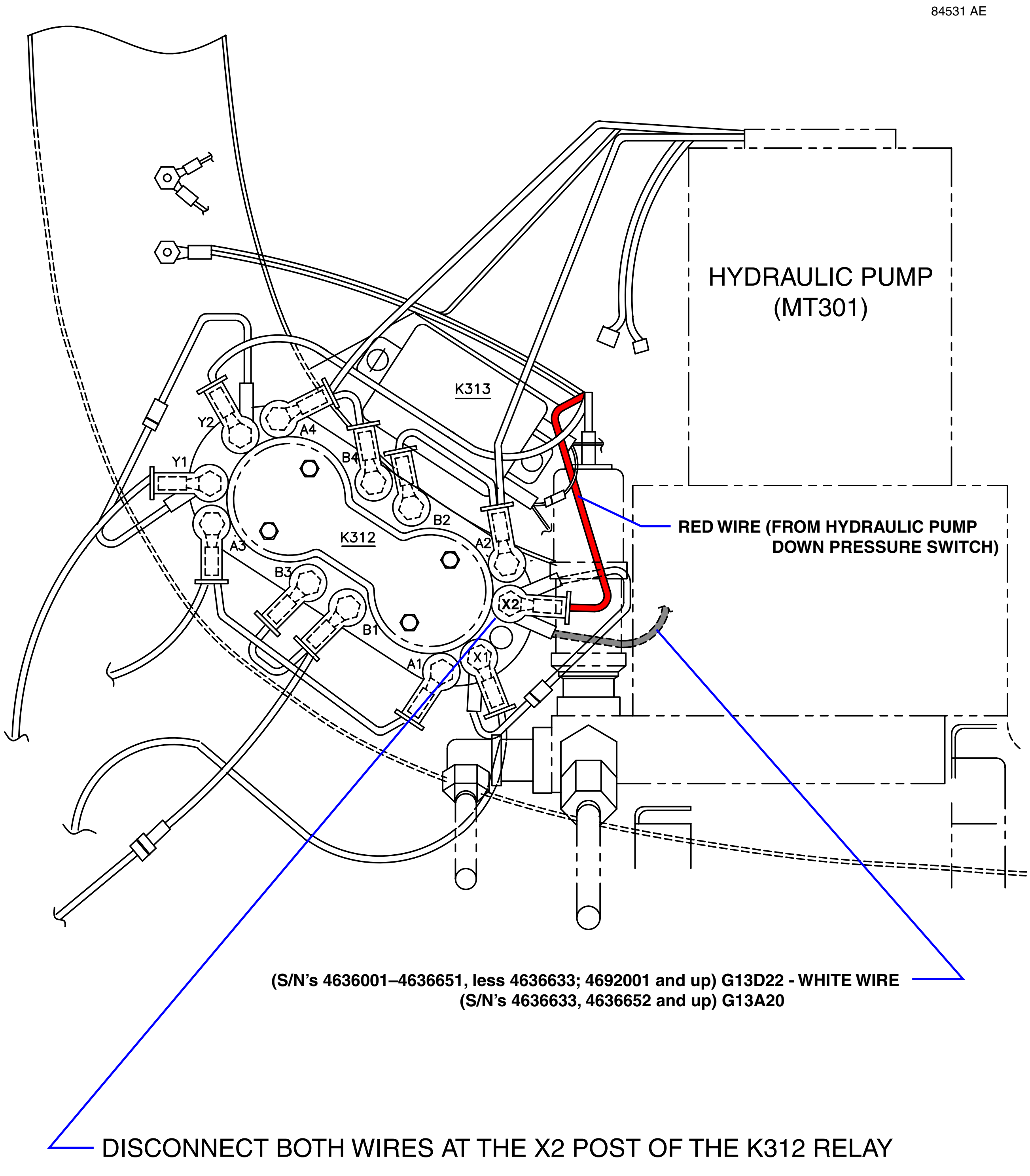
      CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE HYDRAULIC PUMP IS OPERATING. ALWAYS PULL THE HYDRAULIC PUMP CIRCUIT BREAKER BEFORE TESTING THE EMERGENCY GEAR RELEASE SYSTEM. FAILURE TO COMPLY WILL DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

   b) It is acceptable if only the nose gear fully extends down and locked. The main gear may only extend partially. Complete main gear extension by activating emergency gear release valve and moving main gear to the down and locked position.
   c) Reconnect gear down pressure switch circuit. Disconnect gear down pressure switch bypass relay circuit and conduct Gear Selector Switch DOWN Check, below.
   d) Reconnect the gear down pressure switch bypass relay circuit.
2) In S/N's 4636314 and up and S/N's 4692001 and up (i.e. - airplanes equipped with Frisby/Triumph hydraulic pumps only):
   a) Disconnect the gear down pressure switch circuit (i.e., Red Wire, "Figure 1") and conduct the following sequence:
      1] Place the gear selector switch in the UP position. The gear should function normally.
      2] Place the gear selector switch in the DOWN position. The pump will shut OFF before the gears reach the full down position.

         CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE HYDRAULIC PUMP IS OPERATING. ALWAYS PULL THE HYDRAULIC PUMP CIRCUIT BREAKER BEFORE TESTING THE EMERGENCY GEAR RELEASE SYSTEM. FAILURE TO COMPLY WILL DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

         CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB IF ALL THREE GEAR FULLY EXTEND AND LOCK DOWN. DOING SO CAN DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

      It is acceptable if nose gear fully extends down and locked. Main gears may only extend partially. Complete gear extension by pulling emergency gear release knob and moving main gear to the down and locked position.



LOOKING AFT AT RIGHT SIDE OF F.S. 273.746

Down Pressure Switch Bypass Relay
Figure 1

3] Reconnect the gear down pressure switch circuit (i.e., Red Wire, “Figure 1”).

4] Reset the emergency gear release valve.

b) Disconnect the gear down pressure switch by-pass relay circuit (i.e., White Wire, “Figure 1”), and conduct the following sequence:

1] Place the gear selector switch in the UP position. The gear should function normally.

2] Place the gear selector switch in the DOWN position. The gear should function normally.

3] Reconnect the gear down pressure switch by-pass relay circuit (i.e., White Wire, “Figure 1”).

(u) Gear Selector Switch DOWN Check

Place the gear selector switch in the DOWN position and verify that:

1) Hydraulic pump motor operates and the gear extends.

2) All gear return to the down and locked position. Extending fully in:

a) eight (8) seconds or less, in S/N’s 4636001 thru 4636313 (i.e. - airplanes equipped with Parker Hannifin hydraulic pumps only).

b) nine (9) seconds or less, in S/N’s 4636314 and up and S/N’s 4692001 and up (i.e. - airplanes equipped with Frisby/Triumph hydraulic pumps only).

3) Hydraulic pump motor stops operating after the gear is in down and locked position.

CAUTION: ANY MOMENTARY BLINKING OF THE RED OR GREEN GEAR LIGHTS AFTER THE ACTUATOR IS DOWN AND LOCKED MAY INDICATE AN IMPROPERLY ADJUSTED SWITCH IN A LOCKING CYLINDER.

4) Red gear warning light is illuminated while gear is in transit.

5) Three green safe lights are illuminated as the gear locks.

6) Gear warning horn does not sound.

7) Landing light illuminates as the gear door opens.

(v) Turn the landing light switch to the OFF position.

(w) Disconnect the ground power unit.

WARNING: VERIFY GEAR ARE DOWN AND LOCKED BEFORE REMOVING AIRPLANE FROM JACKS.

(x) Remove airplane from the jacks per 7-10-00.

B. S/N's 4636633, 4636652 and Up

(PIR-PPS60033-16, Rev. B)

(1) Setup

(a) Verify that the main and nose gears are properly adjusted (see 32-10-00 and 32-20-00, respectively).

(b) Place the airplane on jacks per 7-10-00.

(c) Connect a 28 Vdc regulated power source capable of supplying a minimum of 50 amperes to the aircraft electrical system.

CAUTION: BEFORE STARTING TEST, CHECK FOR ADEQUATE CLEARANCE OF NOSE GEAR LIGHTS, SEQUENCE VALVES, ACTUATORS, HOSES, ETC.

CAUTION: NEVER OPERATE THE EMERGENCY GEAR RELEASE PISTON WITH THE HYDRAULIC PUMP CIRCUIT BREAKER "IN".

CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE HYDRAULIC PUMP IS OPERATING. ALWAYS PULL THE HYDRAULIC PUMP CIRCUIT BREAKER BEFORE TESTING THE EMERGENCY GEAR RELEASE SYSTEM. FAILURE TO COMPLY WILL DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

NOTE: With power supplied through the auxiliary power plug the batt-master switch should remain in the "OFF" position. The APU circuit bypasses the batt-master switch to energize the ship power circuits.

(d) Set the sequence valve.

1) With the nose gear fully retracted, adjust the plunger screw until the nose gear door actuating torque tube rotates and the sequence lever depresses the plunger approximately 0.125 inch.

2) Lock the plunger and screw in place and connect the nose gear door actuation rods. These rods shall be long initially and shortened progressively until the nose gear doors seat firmly, but are not distorted when closed. Cycle the gear as required to ensure proper operation.

(e) Fluid Level

CAUTION: DO NOT OVERFILL AND DO NOT ALLOW HYDRAULIC FLUID LEVEL TO DROP BELOW THE "L" (LOW) MARK ON THE SEE-THRU RESERVOIR OIL LEVEL DECAL WITH THE GEAR IN THE DOWN POSITION.

During all tests, add clean filtered MIL-PRF-5606 (Revision "F" or higher) hydraulic fluid to the reservoir as necessary to keep proper fluid levels.

1) Verify fluid level is just above the "L" mark on the sight gauge with the landing gear in the down and locked position.

2) Verify that the fluid level does not exceed the "F" mark on the reservoir when the landing gear is in the up and locked position.

(f) Check/Adjust Switches.

1) Check/adjust squat switch per Landing Gear Safety Switch (Squat Switch) Adjustment, 32-60-00.

2) Check to ensure that the gear door closed (gear "UP") switches are wired to the proper panel indicator light (gear warning annunciator).

(g) Bleed the system per 29-10-00, Servicing Hydraulic System.

(2) Procedure

CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE LANDING GEAR ARE DOWN AND LOCKED. DOING SO CAN DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

The following tests and checks shall be performed in the sequence shown. Any failure to respond as specified indicates a malfunction which must be corrected before proceeding.

(a) Avionics Master switch “ON” (remains on throughout all tests).

(b) Enter Maintenance Mode in the Garmin G1000/G1000 NXi system.

NOTE: Maintenance Mode is entered by pressing softkey sequence 1,2,3,1 while on MFD Splash Screen.

NOTE: In airplanes with G1000 NXi only, the MFD Splash Screen does not appear if the system is powered off when the aircraft is ‘airborne’.

(a) This is to shorten the system power up time if power is interrupted during flight – allowing critical Flight Data to be presented to the Pilot more quickly.

(b) Accordingly, the MFD Splash screen may not be available during certain maintenance procedures – which prevents Maintenance Mode from being selected via the MFD softkey press sequence on the Splash Screen.

(c) The MFD Splash Screen will be not available when both of the following conditions are met:

- The aircraft is in ‘airborne’ state (i.e., the Main Gear squat switch is activated) and,
- Power is then removed from the G1000 NXi System.

(d) At the next power up, the MFD Splash Screen will be skipped over.

(e) To recover the MFD Splash Screen, do the following:

- De-activate the Main Gear squat switch and let the airplane sit in the ‘ground’ state for at least one (1) minute, then;
- Removed and reapply power to the aircraft to restart the system.

(f) Maintenance Mode can then be selected from the MFD Splash Screen, before placing the airplane back into ‘airborne’ mode.

(c) Flaps fully retracted (stowed position).

(d) Initial Gear Down Check

NOTE: If the landing gear is not in the “UP” position at the start of this test, place landing gear in the “UP”position before proceeding.

Ensure that the hydraulic pump circuit breaker is “IN”. Place the gear select switch in the “DOWN” position. Move the landing light switch to the “ON” position.

Check the following:

1) Three solid green circles on the EIS strip and EIS reversionary (see “Figure 2” on page 323012 and “Figure 3” on page 323013).

2) Landing light is illuminated.

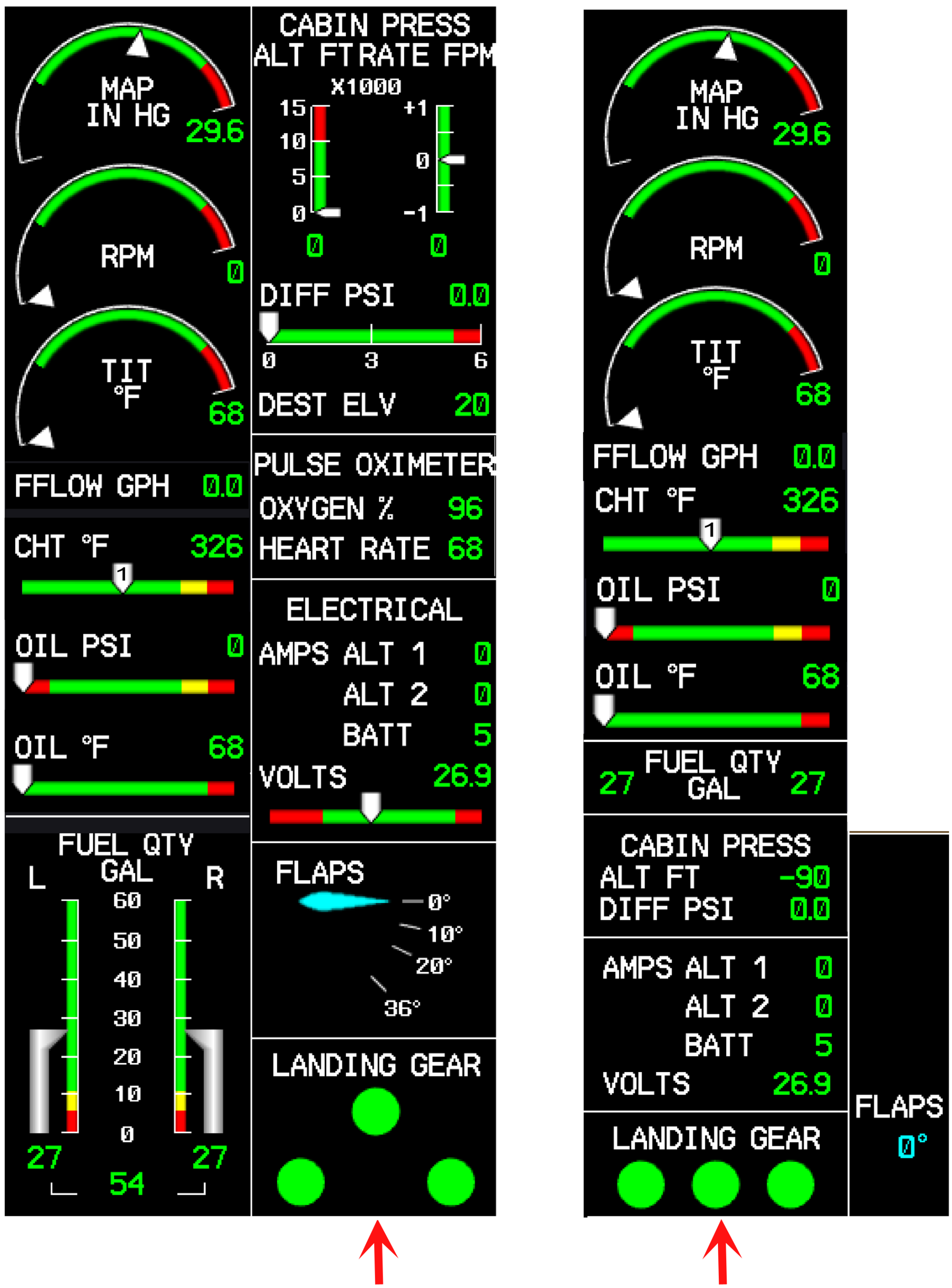
3) Leave the gear down for five minutes and check that the pump motor does not operate at any time. If the pump motor operates at any time during the five minute period, there is a leak or malfunctioning component of the system.

NOTE One momentary pump operation is allowable during this five minute period, provided that gear indicators remain as solid green circles and there is no repeated pump operation for a subsequent fifteen minutes.

EIS STRIPS

NORMAL MODE

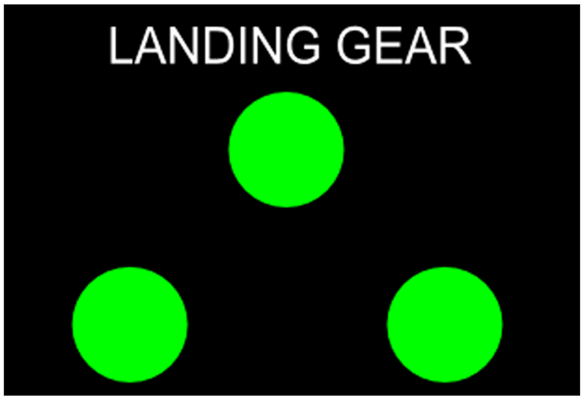
REVERSIONARY MODE



Landing Gear Indication
Figure 2

Effectivity
4636633, 4636652 and up

NORMAL MODE

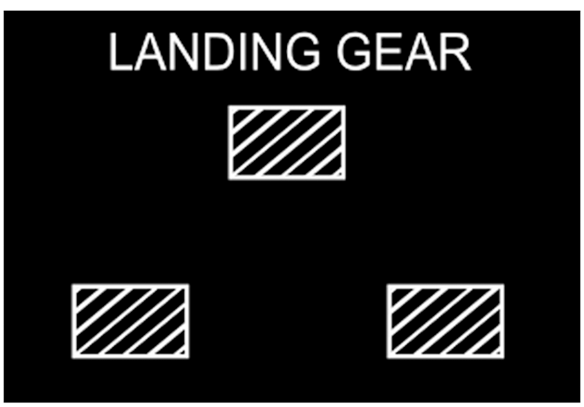


REVERSIONARY MODE

Gear Down and Locked
Figure 3

Effectivity
4636633, 4636652 and up

NORMAL MODE

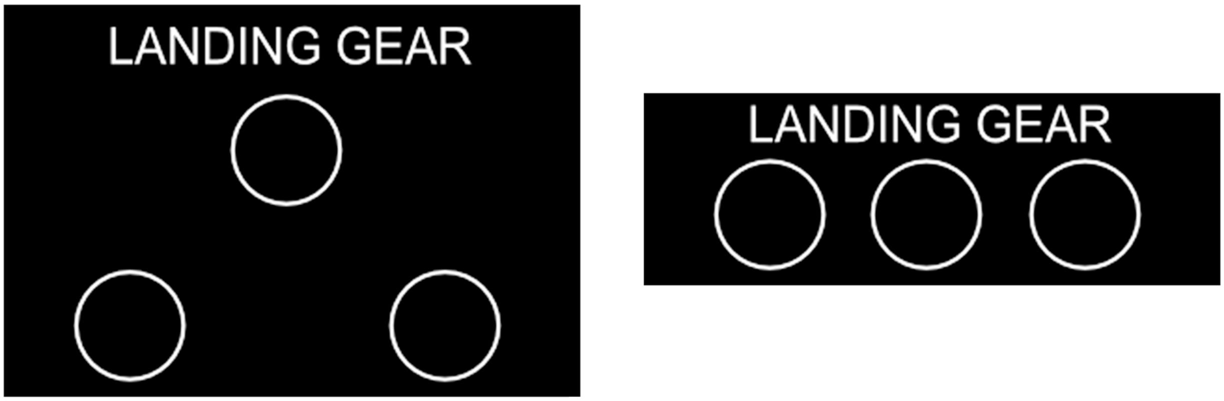


REVERSIONARY MODE

Gear in Transit
Figure 4

Effectivity
4636633, 4636652 and up

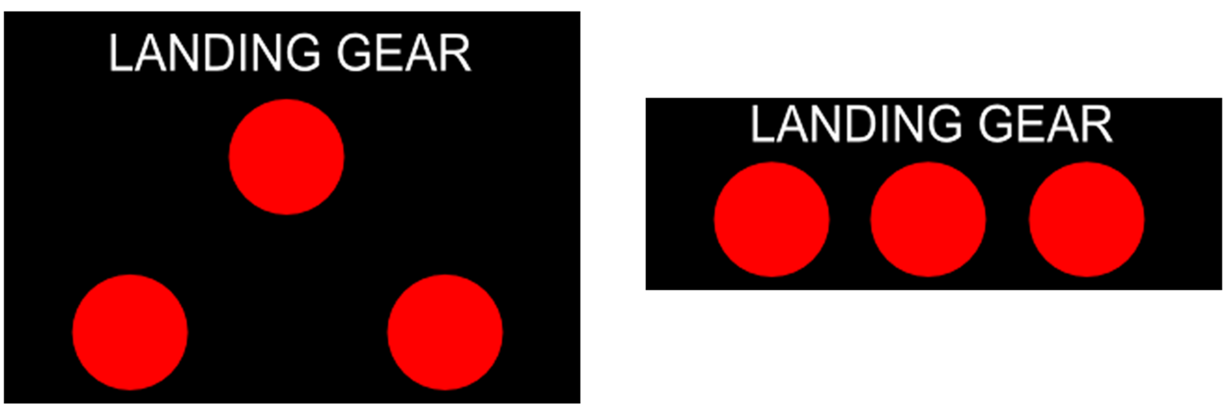
NORMAL MODE **REVERSIONARY MODE**



Effectivity
4636633, 4636652 and up

Gear Up
Figure 5

NORMAL MODE **REVERSIONARY MODE**



Effectivity
4636633, 4636652 and up

Gear Fail
Figure 6

(e) Gear Up Check

Place the gear selector switch in the "UP" position and check the following:

1) While the gear is retracting:

Three rectangles with black/white crosshatch are displayed on the EIS strip and EIS reversionary (see "Figure 4" on page 323013).

2) When the gear is fully retracted:

a) Three white hollow circles are displayed on the EIS strip and EIS reversionary (see "Figure 5" on page 323014).

b) All gear retract fully and the nose gear doors close.

c) Hydraulic pump stops.

d) Landing light is extinguished.

e) Gear retracts fully in 9 seconds or less.

NOTE Any momentary blinking of hollow circles or crosshatched rectangles on the EIS strip after the actuator is in a fully retracted position may indicate an improperly adjusted switch.

3) Leave the gear up for five minutes and check. Pump motor does not operate at any time. (If the pump motor operates at any time during the five minute period, there is a leak in the "UP" line or a malfunctioning component in the system.)

NOTE One momentary pump operation is allowable during this five minute period, provided that gear indicators remain as hollow circles and there is no repeated pump operation for a subsequent fifteen minutes.

(f) Gear Down and Landing Light Check

Put the gear selector switch in the down position and check the following:

While the gear is extending:

1) Three rectangles with black/white crosshatch are displayed on the EIS strip) and EIS reversionary (refer to "Figure 4" on page 323013).

When gear is down and locked:

2) Three solid green circles are displayed on the EIS strip and EIS reversionary (refer to "Figure 2" on page 323012 & "Figure 3" on page 323013)

3) Pump motor stops running.

4) Landing light is illuminated.

5) Move the landing light switch to the "OFF" position.

(g) Gear Free Fall

Move the gear selector switch to the "UP" position.

CAUTION: DO NOT PULL THE EMERGENCY GEAR EXTENSION VALVE HANDLE WHEN THE GEARS ARE DOWN AND LOCKED.

Pull the hydraulic pump circuit breaker "OUT" to disable the pump. Release the guard, pull the gear free fall knob "OUT" and check the following:

1) The gear falls to the down and locked position.

NOTE: The main gear may not fall completely into the locked position during ground free fall tests. Each main gear shall be easily moved into the fully extended position with a side force at centerline of axle not to exceed 9 pounds, and shall then lock in place. Use a force gauge applied at the axle centerline to assure no greater than 9 pounds is applied.

2) Three solid green circles are displayed on the EIS strip and the EIS reversionary (see “Figure 2” on page 323012 & “Figure 3” on page 323013).

3) “GEAR SYS CAS” message is displayed on the PFD.

4) Allow pressure to bleed off befor continuing (apporoximately 2 minutes).

(h) Gear Free Fall Reset

1) Place the gear selector switch in the “DOWN” position, push the free fall knob in and engage guard.

2) Push the “HYDRAULIC PUMP” circuit breaker in and check the following:

3) The three solid green circles remain displayed (see “Figure 2” on page 323012 & “Figure 3” on page 323013.)

4) The hydraulic pump may run momentarily.

5) Cycle gear up & down one time to ensure normal operation.

(i) Gear Down Failure

1) Move the gear selector switch to the “UP” position.

2) Pull the hydraulic pump circuit breaker “OUT” mid stroke.

After 16 seconds check the following:

a) Three solid red circles are displayed (see “Figure 6” on page 323014).

b) Aural Master Warning Tone is sounding.

NOTE: Landing Gear on EIS will flash red until acknowledged.

NOTE: Timing begins when the gear selector is in the “UP” position.

3) Acknowledge failure by depressing the “MASTER WARNING/CAUTION” soft key.

Check the following:

a) The three solid red circles remain displayed (see “Figure 6” on page 323014).

b) Aural Master Warning Tone stops sounding.

4) Put the hydraulic pump circuit breaker “IN”.

Check the following:

a) Hydraulic pump motor operates.

b) When the hydraulic pump stops running, three white hollow circles are displayed (see “Figure 5” on page 323014).

(j) Gear Down Pressure Switch By-Pass Circuit Test

1) Disconnect the red wire from the K312 X2 post (hydraulic pump gear down pressure switch circuit) (see “Figure 1” on page 32308) and perform the following in sequence:

a) Place the gear selector switch in the “UP” position. The gear should function normally.

b) Place the gear selector switch in the “DOWN” position. The pump will shut “OFF” before the gears reach the full down position. The nose gear should fully extend down and lock. The main gears may only extend partially.

c) Complete gear extension by activating the emergency gear release valve and moving the main gear to the down and locked position.

d) Reconnect the red wire to the K312 X2 post (hydraulic pump gear down pressure switch circuit).

e) Reset the emergency gear release valve.

2) Disconnect the white G13A20 wire from the K312 X2 post (hydraulic pump gear down pressure switch circuit) (see "Figure 1" on page 32308) and conduct the following in sequence:

a) Place the gear selector switch in the "UP" position. The gear should function normally.

b) Place the gear selector switch in the "DOWN" position. The gear should function normally.

c) Reconnect the white G13A20 wire to the K312 X2 post (hydraulic pump gear down pressure switch circuit).

(k) Completion of Test

1) Move the gear selector switch to the "DOWN" position and wait for the gear to fully extend, with three solid green circles displayed (see "Figure 2" on page 323012 & "Figure 3" on page 323013).

2) Ensure that the squat switch is still adjusted per section 32-60-00

3) Avionics Master switch "OFF". Disconnect external power receptacle from airplane.

WARNING: VERIFY GEAR ARE DOWN AND LOCKED BEFORE REMOVING AIRPLANE FROM JACKS.

4) Remove airplane from jacks.

3. Nose Gear Actuator Assembly

See "Figure 7" on page 323018.

NOTE: No field repair of landing gear actuators installed in these airplanes is authorized. Actuator overhaul service is available through Piper dealers.

A. Removal

(1) Place airplane on jacks (refer to 7-10-00).

(2) Reduce hydraulic pressure as follows:

(a) Place the gear control in the UP position.

(b) Allow gear to retract approximately halfway and pull Hydraulic Pump circuit breaker.

NOTE: Have an assistant stand by nose gear to catch the gear and prevent it from falling back to a down-and-locked position. Block the nose gear to hold that position.

(c) Set the BATT MASTER switch to the OFF position.

CAUTION: DO NOT ATTEMPT TO REMOVE OR DISCONNECT EITHER END OF THE NOSE GEAR DOWNSPRING ASSEMBLY WITHOUT THE USE OF THE SPECIAL TOOL (P/N 762-157) SHOWN IN "FIGURE 2" ON PAGE 32206. THE DOWNSPRING ASSEMBLY IS HEAVILY LOADED.

CAUTION: ONCE THE REMOVE/INSTALL TOOL (P/N 762-157) IS IN PLACE, USE HOSE CLAMPS TO SECURE IT TO THE DOWNSPRING ASSEMBLY.

(3) Position the special remove/install tool (P/N 762-157) (see "Figure 2" on page 32206) so that the nose gear downspring is within the two end plates. Secure the special tool to the downspring assembly with hose clamps. Remove bolt holding actuator rod end bearing and downspring assembly to trunnion. Remove washers, spacers, and bushing.

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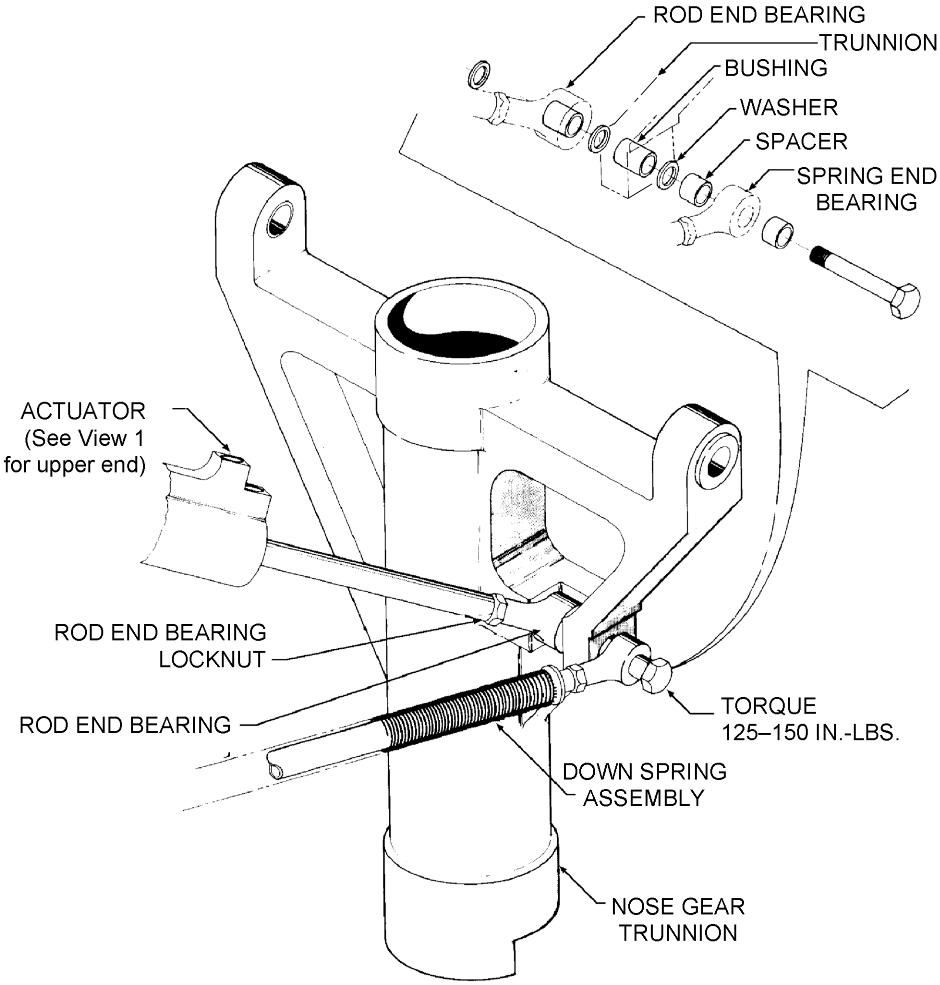
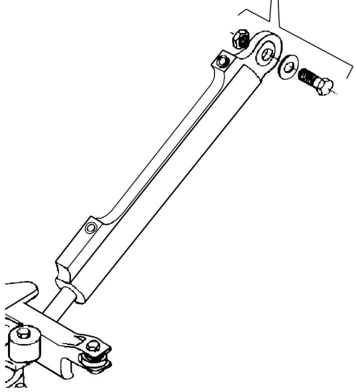


S/N's 4636001–4636620 and 4692001–4692207 with engine mount P/N 89137-041 or P/N 89137-042 installed.

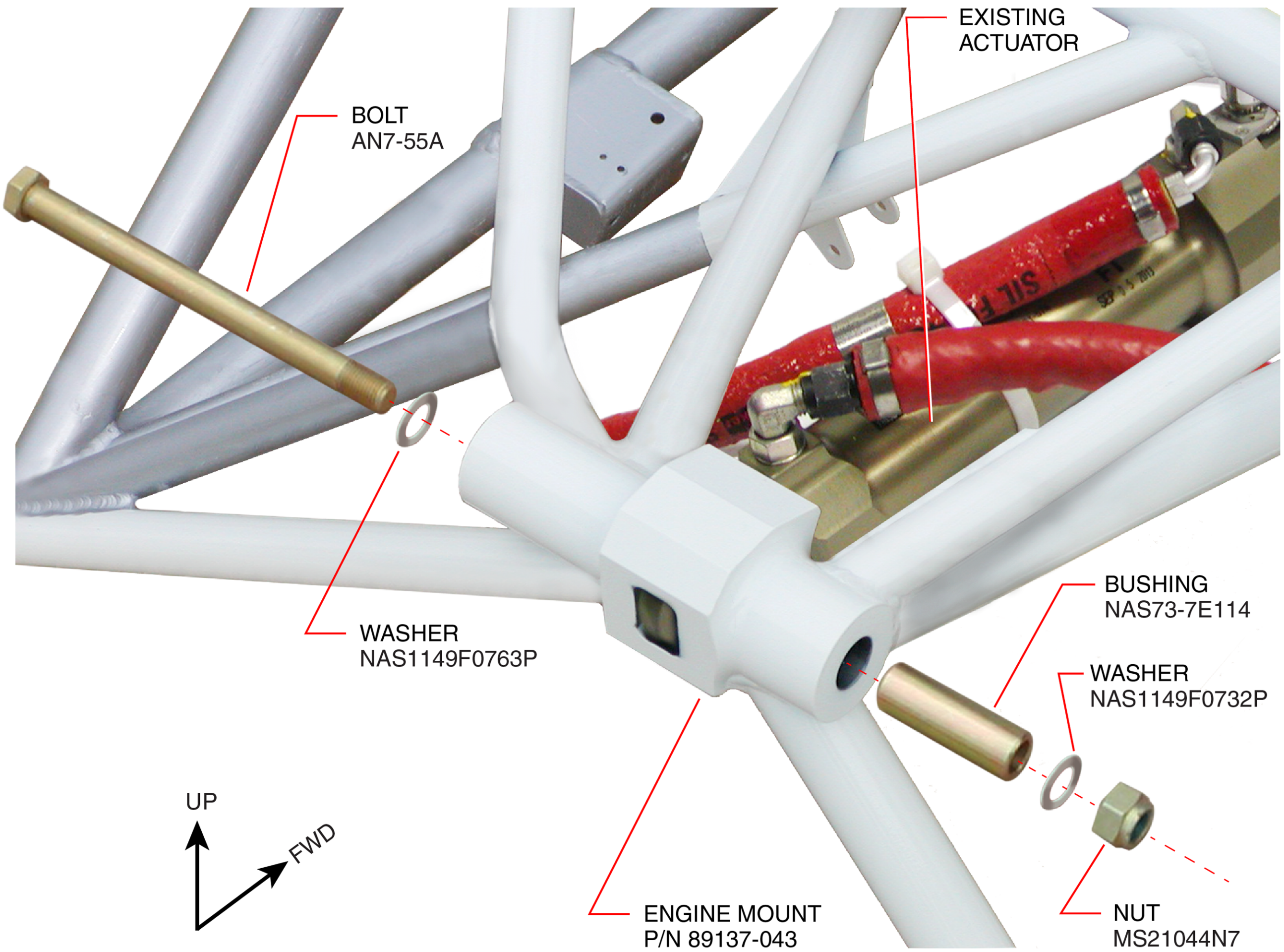
VIEW 1



NOTE: It is acceptable to use NAS1149F0763P washer in place of NAS1149F0732P washer to provide additional clearance between the actuator assembly and the engine mount tube.

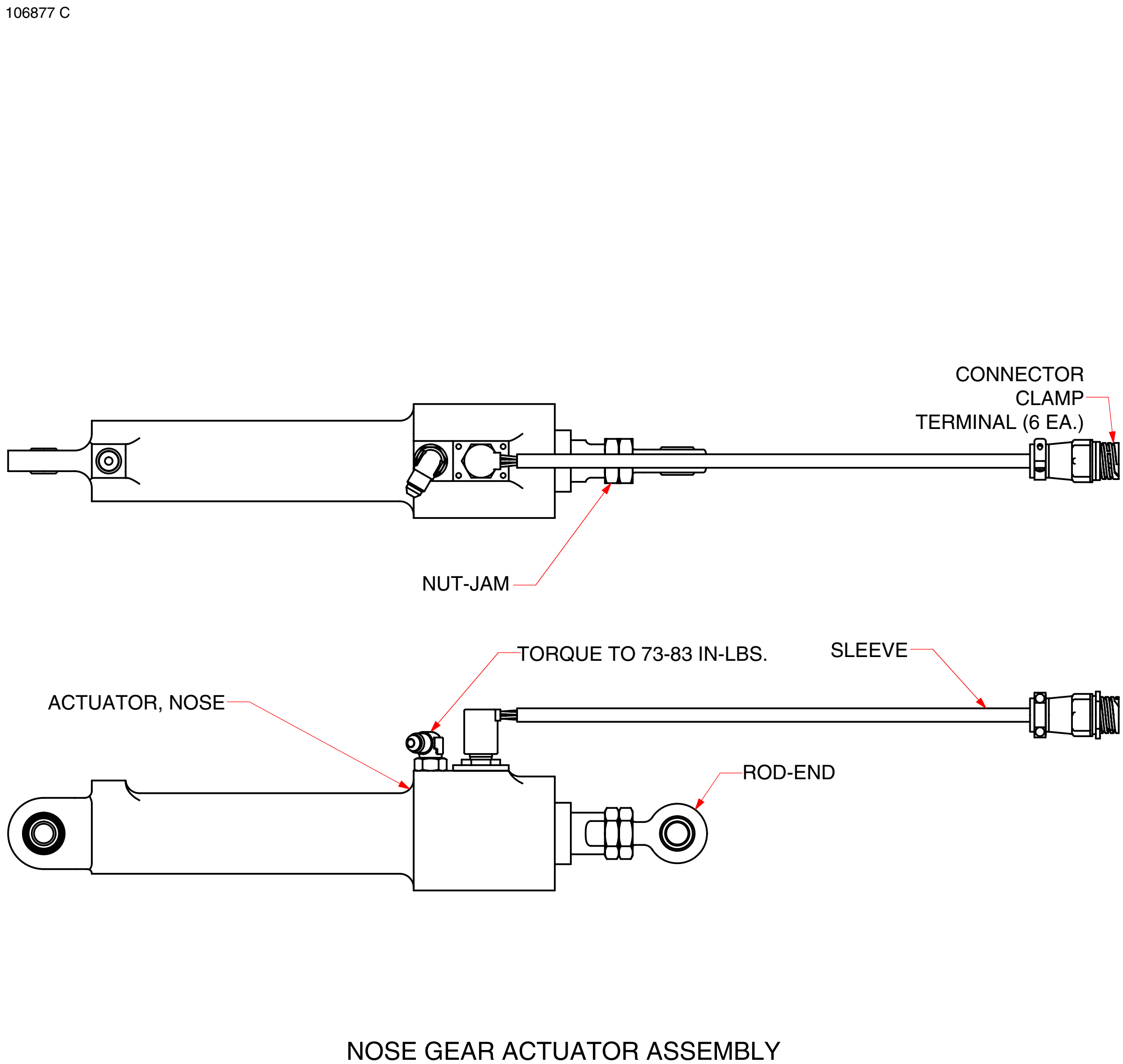
Nose Gear Actuator Installation
Figure 7 (Sheet 1 of 4)

VIEW 1



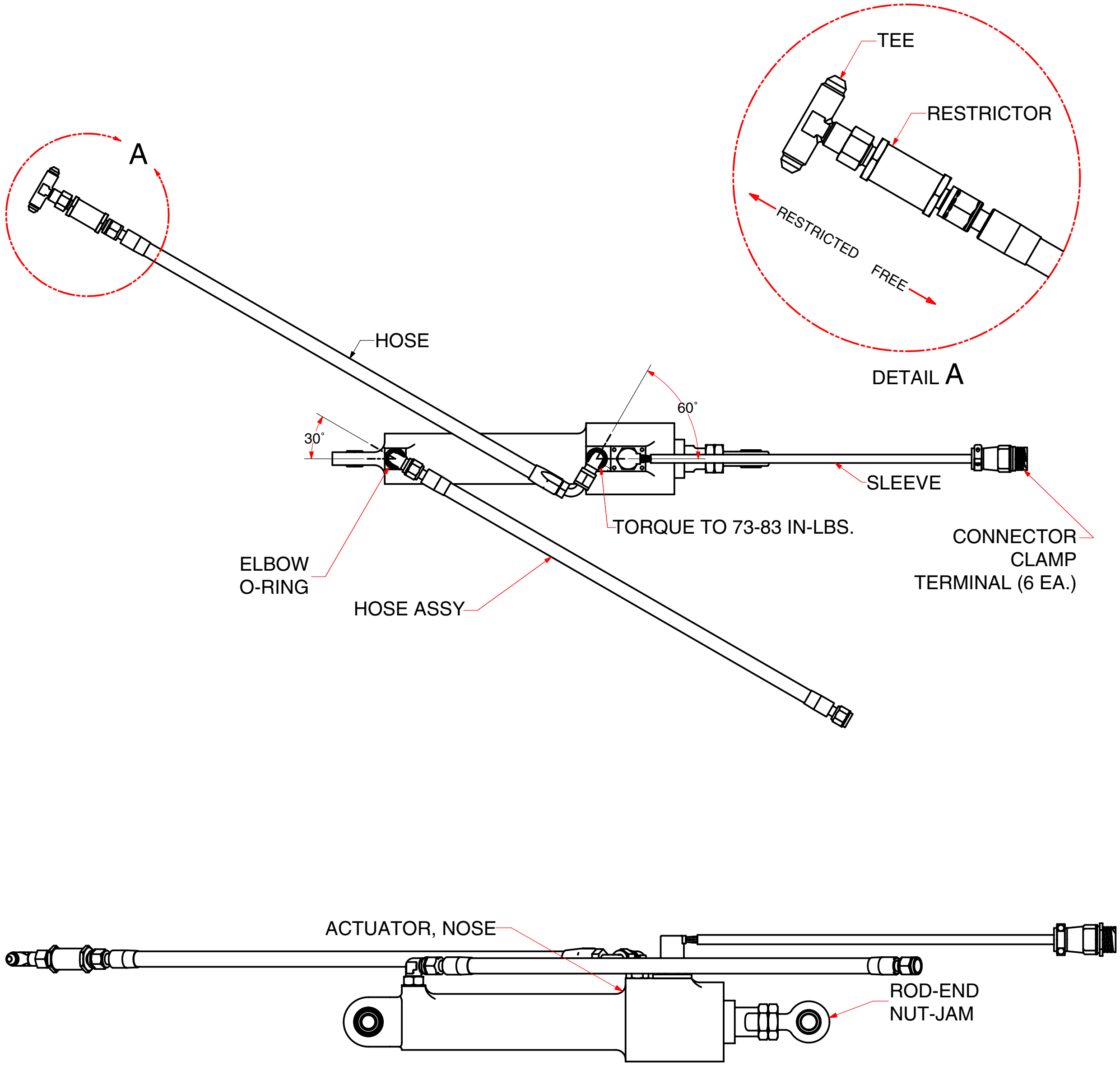
Nose Gear Actuator Installation
Figure 7 (Sheet 2 of 4)

Effectivity
with P/N 89137-043 installed



FACTORY INSTALLED: S/N's 4692001 and up.
S/N's 4636001–4636716.

Nose Gear Actuator Installation
Figure 7 (Sheet 3 of 4)



NOSE GEAR ACTUATOR ASSEMBLY

FACTORY INSTALLED: S/N's 4636717 and up.

AUTHORIZED SERVICE REPLACEMENT: S/N's 4636001–4636716.
S/N's 4692001 and up.

Nose Gear Actuator Installation
Figure 7 (Sheet 4 of 4)

CAUTION: IF COMPLETE DISASSEMBLY OF THE DOWNSPRING MECHANISM IS DESIRED, USE OF THE SPECIAL DISASSEMBLY/REASSEMBLY TOOL (P/N 762-155) IS REQUIRED. THE EMERGENCY DOWNSPRING MECHANISM IS SPRING LOADED AND COULD CAUSE SERIOUS INJURY OR DAMAGE IF THE SPECIAL DISASSEMBLY/REASSEMBLY TOOL (P/N 762-155) IS NOT USED.

(4) Disconnect electrical leads from downlimit micro-switch.

(5) Disconnect and quickly cap two hydraulic lines at nose gear actuator. Remove safety wire, if installed.

NOTE: If the hydraulic line elbow on the lower end of the nose gear actuator is to be removed (i.e. - if replacing actuator), it may be necessary to first remove the downlimit micro-switch. If so, remove only the four screws holding the switch plate to the actuator. DO NOT move or adjust the jam nuts and lockwire attaching the micro-switch to the switch plate as that would affect lock engagement verification.

(6) Move landing gear selector to the DOWN position.

(7) Manually extend rod so that actuator assembly locks in down, or fully extended, position.

(8) Measure and record distance from end fitting to center of rod end bearing. This measurement is important for proper installation.

WARNING: IF EITHER THE NOSE GEAR OR ACTUATOR IS DAMAGED OR MISALIGNED, THIS MEASUREMENT MAY NOT BE VALID. IN EITHER CASE, REPLACE DAMAGED COMPONENTS, AS REQUIRED, AND USE MALIBU NOSE GEAR ACTUATOR ALIGNMENT TOOL (P/N 762-137) DURING INSTALLATION.

(9) Remove rod end bearing from actuator rod. Check that rod end bearing locknut is free of scoring and nicks. Smooth out if necessary.

(10) Remove actuator from engine mount.

B. Installation

See “Figure 8” on page 324015.

NOTE: If the downlimit micro-switch was previously removed from the actuator, install it and tighten the four screws in a criss-cross pattern, torquing each 13 to 17 in. lbs. Verify lock indication in the extended position and unlock indication in the retracted position with a multi-meter or continuity tester.

(1) Purge actuator per “Component Purge Procedures” on page 323038.

(2) Manually extend rod so that actuator assembly locks in down, or fully extended, position.

(3) Install rod end bearing in actuator rod. Adjust to same length as recorded during Removal, see Step “(8)” above. (See WARNING following Step “(8)” above.)

(4) Install upper end of actuator to engine mount.

NOTE: Pay close attention to the nose gear actuator / engine mount connecting hardware to ensure the correct hardware is used per “Figure 8” on page 324015. Additionally, check for proper thread engagement between the bolt and nut by ensuring at least one thread of the bolt protrudes through the nut per AC 43.13-1, latest revision. If there are not sufficient threads exposed after the lock nut, replace the bolt with a longer bolt, i. e., the next longer in the same bolt series.

(5) Uncap and connect hydraulic lines to actuator.

NOTE: If both actuator and hose end fitting are drilled for safety wire, install safety wire. See Piper Service Letter 1202, latest revision.

(6) Connect electrical leads to downlimit micro-switch.

(7) Unlock actuator as follows:

(a) Set the BATT MASTER switch to the ON position.

NOTE: Have an assistant stand by the actuator to observe the rod position and signal as soon as the rod retracts from the fully extended and locked position.

(b) Place the gear control in the UP position and when signalled by the assistant, pull the Hydraulic Pump circuit breaker.

(c) Set the BATT MASTER switch to the OFF position.

(8) Install rod end bearing and nose gear down assist spring in trunnion. Torque per “Figure 8” on page 324015.

(9) Check adjustment and alignment of nose gear per 32-20-00.

(10) Tighten rod end bearing locknut.

NOTE: If unable to rig nose wheel assembly to proper retracted location with two jam nuts installed, remove one jam nut. Maximum thread exposure with single nut is 0.28 inch.

(11) Purge air from system per Bleeding Hydraulic System, 29-10-00.

(12) Check that hydraulic pump circuit breaker is on (pressed in), gear control lever is in the DOWN position, and BATT MASTER switch is in the OFF position.

(13) Set BATT MASTER switch to the ON position. Cycle landing gear up and down at least three (3) times. Watch that the gear operates smoothly and check that there are no hydraulic leaks.

(14) Service reservoir as required.

(15) Remove airplane from jacks (see 7-10-00).

4. Nose Gear Down Spring Assembly

A. Removal

(1) Remove the engine cowling. (Refer to 71-10-00.)

(2) Place the airplane on jacks. (Refer to 7-10-00.)

(3) Retract the nose gear slightly to remove it from the down locked position.

WARNING: USE HOSE CLAMPS TO HOLD THE DOWN SPRING WITHIN THE CONFINES OF SPECIAL TOOL. DO NOT ATTEMPT TO RELEASE DOWN SPRING WITHOUT THE USE OF THE SPECIAL TOOL. THE EMERGENCY DOWNSPRING MECHANISM IS SPRING LOADED AND COULD CAUSE SERIOUS INJURY OR DAMAGE IF THE SPECIAL DISASSEMBLY/REASSEMBLY TOOL (P/N 762-155) IS NOT USED.

(4) Use the special tool Piper P/N 762-157 (see “Figure 2” on page 32206) to hold the down spring at at its compressed length for installation.

(5) Disconnect the actuator and emergency down spring assembly from the right-hand side of the trunnion by removing the safety wire, bolt, spacers and washers. See “Figure 7” on page 323018.

(6) Attach a lifting sling to the engine suspension points and using a one-half ton hoist, support the engine.

(7) Disconnect the upper rod end of the down spring assembly from the right lower engine mount attachement point by removing the nut, washers, and bolt. See 71-20-00 paragraph 1C.

(8) Remove the down spring assembly.

B. Installation

NOTE: The following assumes the down spring assembly is compressed and the spring is captured by the down spring retaining tool. If not, see ASSEMBLY below.

(1) Position the upper rod end of the down spring assembly in place at the right lower engine mount attachement point and secure the rod end, fuselage mounting bracket and engine mount with bolt, washers, and nut. Torque as specified. See 71-20-00 paragraph 1C.

(2) Remove the lifting sling and hoist supporting the engine.

(3) Position the actuator and emergency down spring assembly lower rod end in place at the right-hand side of the nose gear trunnion. Secure with bolt, spacers and washerand safety.

(4) Remove the down spring retaining tool.

(5) Cycle the gear a minimum of three times to ensure proper operation.

(6) Ascertain that gear is lubricated per 12-20-00, Chart 4.

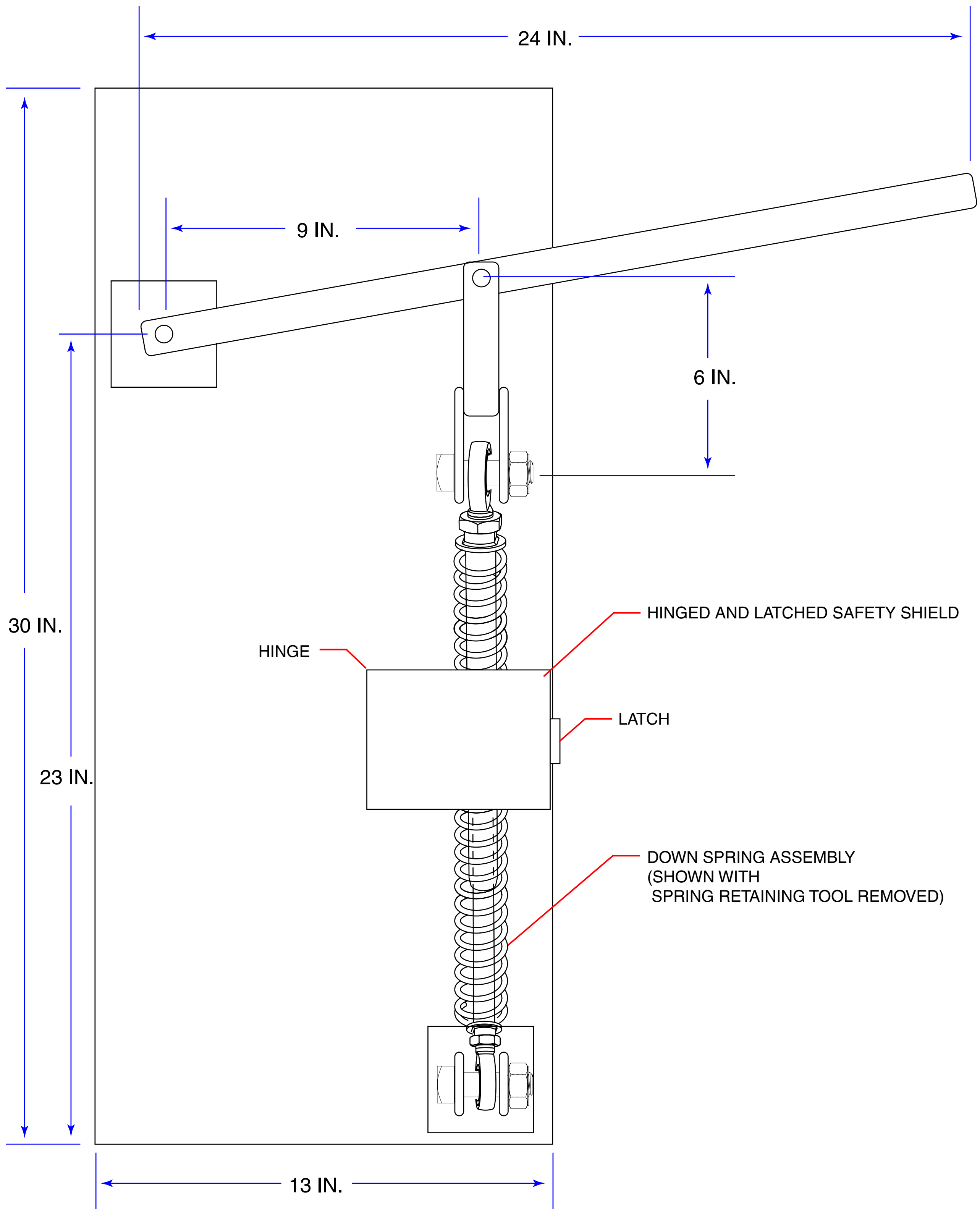
(7) Install engine cowling. (Refer to 71-10-00.)

C. Disassembly

(1) Using a suitable tool/fixture (see “Figure 8” on page 323025 for a suggested configuration), compress the down spring assembly sufficient to allow the hose clamps and down spring retaining tool to be safely removed.

(2) Carefully allow the down spring assembly to extend to the full uncompressed length of the spring.

(3) The down spring assembly can now be disassembled simply by pulling the components apart.



Down Spring Disassembly Tool
Figure 8

D. Assembly

(1) Reassemble the down sprIng assembly as follows:

(a) Place the down spring over the lower rod assembly.

(b) Insert the upper rod assembly into the down spring and lower rod assembly.

(2) Place the assembled down spring assembly into a suitable tool/fixture (see “Figure 8” on page 323025 for a suggested configuration).

WARNING: USE HOSE CLAMPS TO HOLD THE DOWN SPRING WITHIN THE CONFINES OF SPECIAL TOOL. DO NOT ATTEMPT TO RELEASE DOWN SPRING WITHOUT THE USE OF THE SPECIAL TOOL. THE EMERGENCY DOWNSPRING MECHANISM IS SPRING LOADED AND COULD CAUSE SERIOUS INJURY OR DAMAGE IF THE SPECIAL DISASSEMBLY/REASSEMBLY TOOL (P/N 762-155) IS NOT USED.

(3) Carefully compress the down spring assembly sufficient to install the special tool Piper P/N 762-157 (see “Figure 2” on page 32206) to hold the down spring at proper length for installation. Secure with hose clamps.

(4) Proceed with “Installation” on page 323024.

5. Main Gear Actuator Assembly

NOTE: No field repair of landing gear actuators installed in these airplanes is authorized. Actuator overhaul service is available through Piper dealers.

A. Removal

(1) Place airplane on jacks per 7-10-00.

(2) Reduce hydraulic pressure as follows:

(a) Place gear control in the UP position.

(b) Allow gear to retract approximately halfway and pull Hydraulic Pump circuit breaker.

NOTE: Have assistants standing by each main gear to catch the gear and prevent them from falling back to a down and locked position. Block each main gear to hold that position.

(c) Set BATT MASTER switch to the OFF position.

(3) Disconnect and quickly cap hydraulic lines at main gear actuator. Remove safety wire, if installed.

(4) Disconnect actuator from landing gear trunnion by removing washer, bushing, and bolt.

(5) Disconnect upper end of actuator from its mounting bracket by removing bolt and washer.

(6) Remove actuator from airplane.

(7) Remove rod end bearing from actuator rod. Check that rod end bearing lock nut is free of scoring and nicks. Smooth out if necessary.

(8) Move landing gear selector to the DOWN position.

(9) Manually extend rod so that actuator assembly locks in down, or fully extended position.

B. Installation

NOTE: In S/N's 4636001 thru 4636195 only, when installing service replacement actuators P/N's:

89075-015, 89075-013, 89075-020, or 106877-002 (RH)
89075-016, 89075-012, 89075-019, or 106877-002 (LH)

for the first time, modify the associated original equipment main gear trunnion per Trunnion Modification, below.

(1) Purge actuator per "Component Purge Procedures" on page 323038.

(2) Install rod end bearing on actuator rod.

(3) Bolt upper end of actuator to its mounting bracket. Torque to 210-270 in.-lbs. Safety the bolt.

(4) Position gear in order to bolt rod end bearing to main gear trunnion. Torque to 260-320 in.-lbs. Safety the bolt.

(5) Check adjustment of rod end bearing per 32-10-00, Main Gear Assembly, Adjustment.

(6) Tighten rod end bearing locknut.

(7) Uncap and connect hydraulic lines.

NOTE: If both actuator and hose end fitting are drilled for safety wire, install safety wire. See Piper Service Letter 1202, latest revision.

(8) Purge system of air per Bleeding Hydraulic System, 29-10-00.

(9) Check that hydraulic pump circuit breaker is set (pressed in), gear control switch is in the DOWN position and BATT MASTER switch is OFF.

(10) Remove airplane from jacks (see 7-10-00).

C. Trunnion Modification (S/N's 4636001 thru 4636195 only.)

NOTE: This modification is equivilent to Piper Kit No. 88275.

Service replacement actuators incorporate a larger rod end bearing than the actuators factory installed in S/N's 4636001 thru 4636195. This larger rod end bearing will not adequately clear the original equipment main gear trunnion. Accordingly, the first time either original equipment main gear actuator is replaced with any of the 89075-015, 89075-013, 89075-020, or 106877-002 (RH); or 89075-016, 89075-012, 89075-019, or 106877-002 (LH) service replacement actuators, modify the associated original equipment trunnion as follows:

(1) Remove main gear actuator per Removal, above.

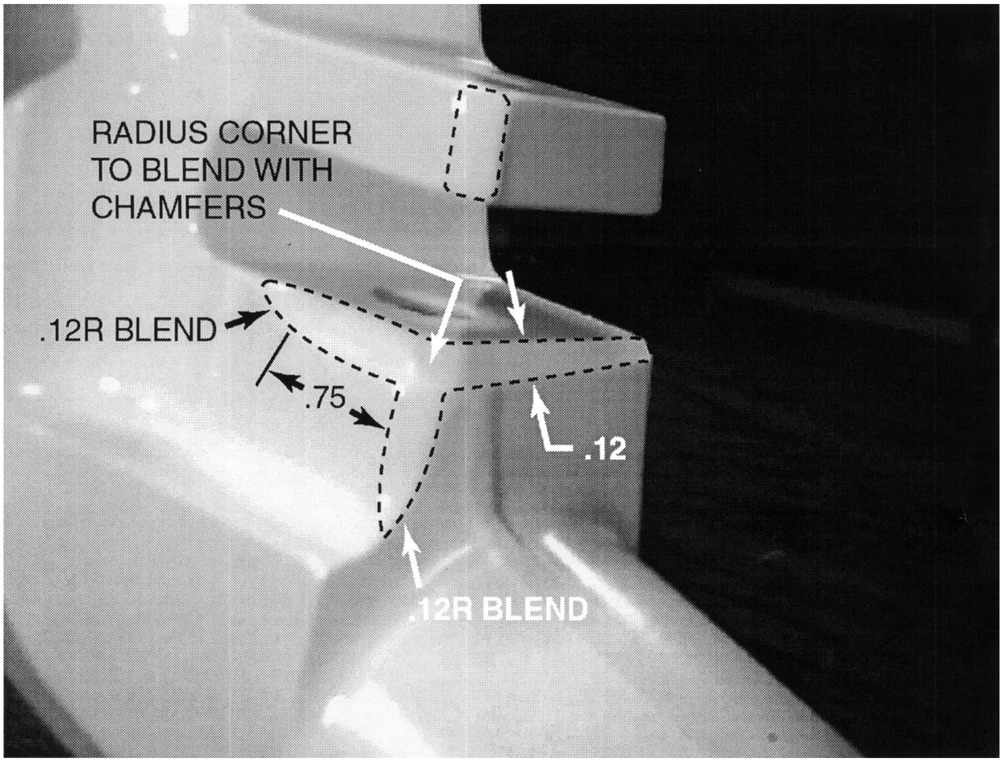
(2) Chamfer the edges of the main gear trunnion as specified in "Figure 9" on page 323029.

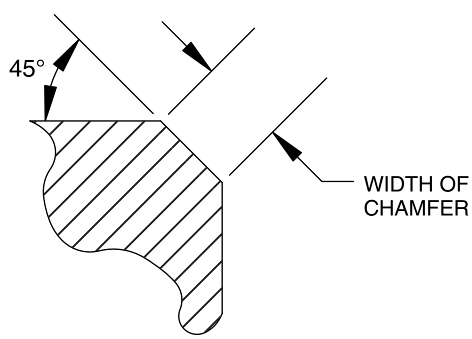
(3) Re-install the actuator per Installation, above.

(4) Swing the gear and observe to assess rod end/trunnion clearance. If required, repeat steps (1) - (3), above, adjusting the depth of the chamfer as required to obtain adequate clearance for normal gear function.

(5) When adequate clearance has been obtained for normal gear function, prime and paint reworked area to match the remaining trunnion finish.

(6) Make log book entry for completion of this modification.





THE CHAMFER WIDTH IS TYPICALLY .25 INCHES
EXCEPT FOR THE .12 INCH CHAMFER WIDTH
SHOWN IN THE PHOTO.

Chamfer Details
Figure 9

6. Nose Gear Door Actuator Assembly

See “Figure 10”.

NOTE: No field repair of nose gear door actuators installed in these airplanes is authorized. Actuator overhaul service is available through Piper dealers.

A. Removal

(1) Place airplane on jacks (refer to 7-10-00).

(2) Reduce hydraulic pressure as follows:

(a) Place gear control in the UP position.

(b) Allow gear to retract approximately halfway and pull Hydraulic Pump circuit breaker.

NOTE: Have an assistant stand by nose gear to catch the gear and prevent it from falling back to a down and locked position. Block the nose gear to hold that position.

(c) Set BATT MASTER switch to the OFF position.

(3) Disconnect and quickly cap hydraulic line to nose gear door actuator. Tie capped line to airframe.

(4) Remove bolt, nut, and washers holding rod end to nose gear/tube assy.

(5) Remove clevis bolt, four washers, nut, and bushing from aft end of actuator.

(6) Remove actuator from airplane.

B. Installation

(1) Purge actuator per “Component Purge Procedures” on page 323038.

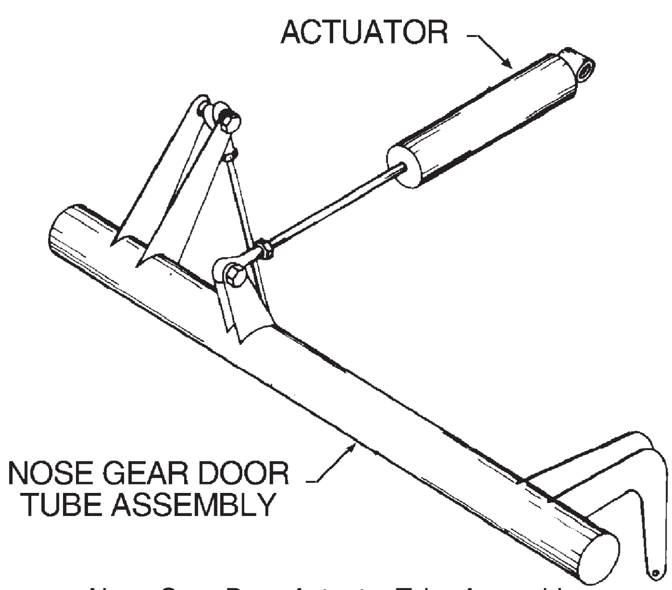
(2) Position actuator in airplane and secure aft end of actuator with clevis bolt, four washers, nut, and bushing.

(3) Bolt rod end to nose gear door tube assy with bolt, nut, and two washers.

(4) Uncap and connect hydraulic line to nose gear door actuator. Purge system of air per Bleeding Hydraulic System, 29-10-00.

(5) Check that hydraulic pump circuit breaker is set (pressed in), gear control switch is in the DOWN position and BATT MASTER switch is OFF.

(6) Remove airplane from jacks (see 7-10-00.)



Nose Gear Door Actuator Tube Assembly
Figure 10

7. Sequence Valve Parts Replacement (Parker Hannifin Only)

See “Figure 11” on page 323032 and “Figure 12” on page 323033.

CAUTION: TAKE CARE NOT TO LOSE ANY SMALL PARTS IN VALVE.

A. Drain valve by removing two plugs. Remove and discard O-rings.

B. Remove retaining ring from restricter port in valve body.

C. Remove plug at restricter port using three-quarter-inch screw with 4-40 threads.

D. Remove and discard O-ring.

E. Remove spring and restricter. Tap valve at restricter port into palm of hand to aid removal of restricter.

F. Remove retaining ring from plunger port. Remove plug using three-quarter inch screw with 4-40 threads.

G. Remove spring, spring retainer, and ball from plunger port.

H. Remove and discard O-ring from plunger port.

I. Remove bolt and locknut to remove plunger, seat, and spring. Use a No. 1 Phillips screwdriver to aid in removal of plunger.

J. Remove and discard O-ring from seat.

K. Remove and discard O-ring from plunger port.

WARNING: USE SOLVENT IN WELL VENTILATED AREA. AVOID CONTACT WITH SKIN. KEEP AWAY FROM HEAT OR FLAME.

L. Clean parts as needed with solvent.

CAUTION: DO NOT REUSE O-RINGS. COAT REPLACEMENT O-RING WITH HYDRAULIC FLUID MIL-PRF-5606.

M. Lubricate all ports with hydraulic fluid.

N. Insert restricter and spring. Install O-ring onto plug and install plug. Secure with retaining ring.

O. Install O-ring, plunger, and spring into valve body. Install O-ring onto seat and install seat.

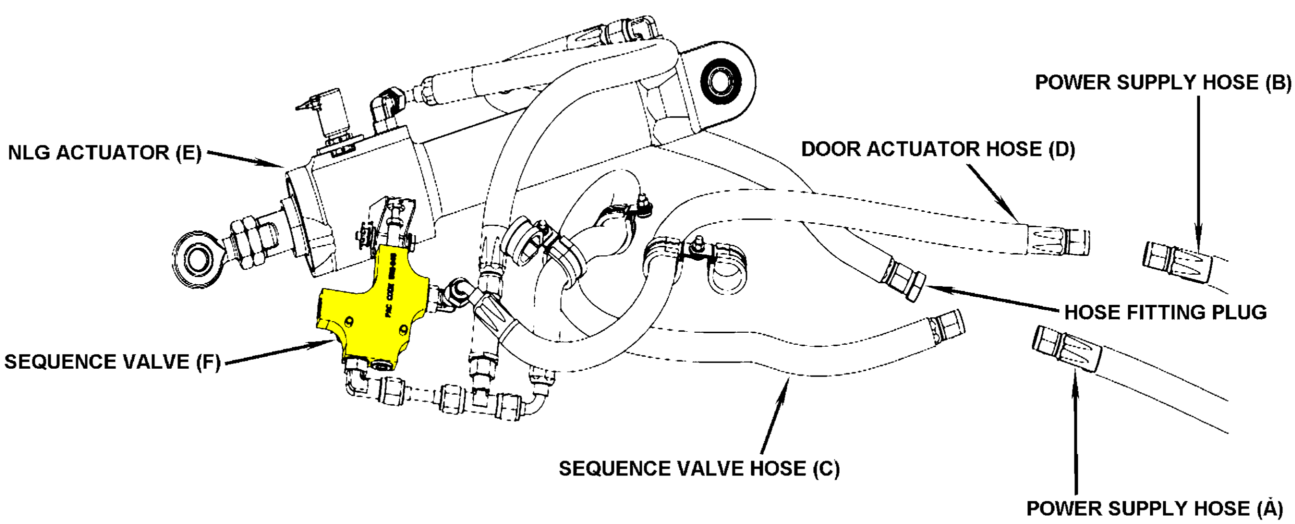
P. Install O-ring, ball, spring retainer, and spring.

Q. Insert plug and retaining ring.

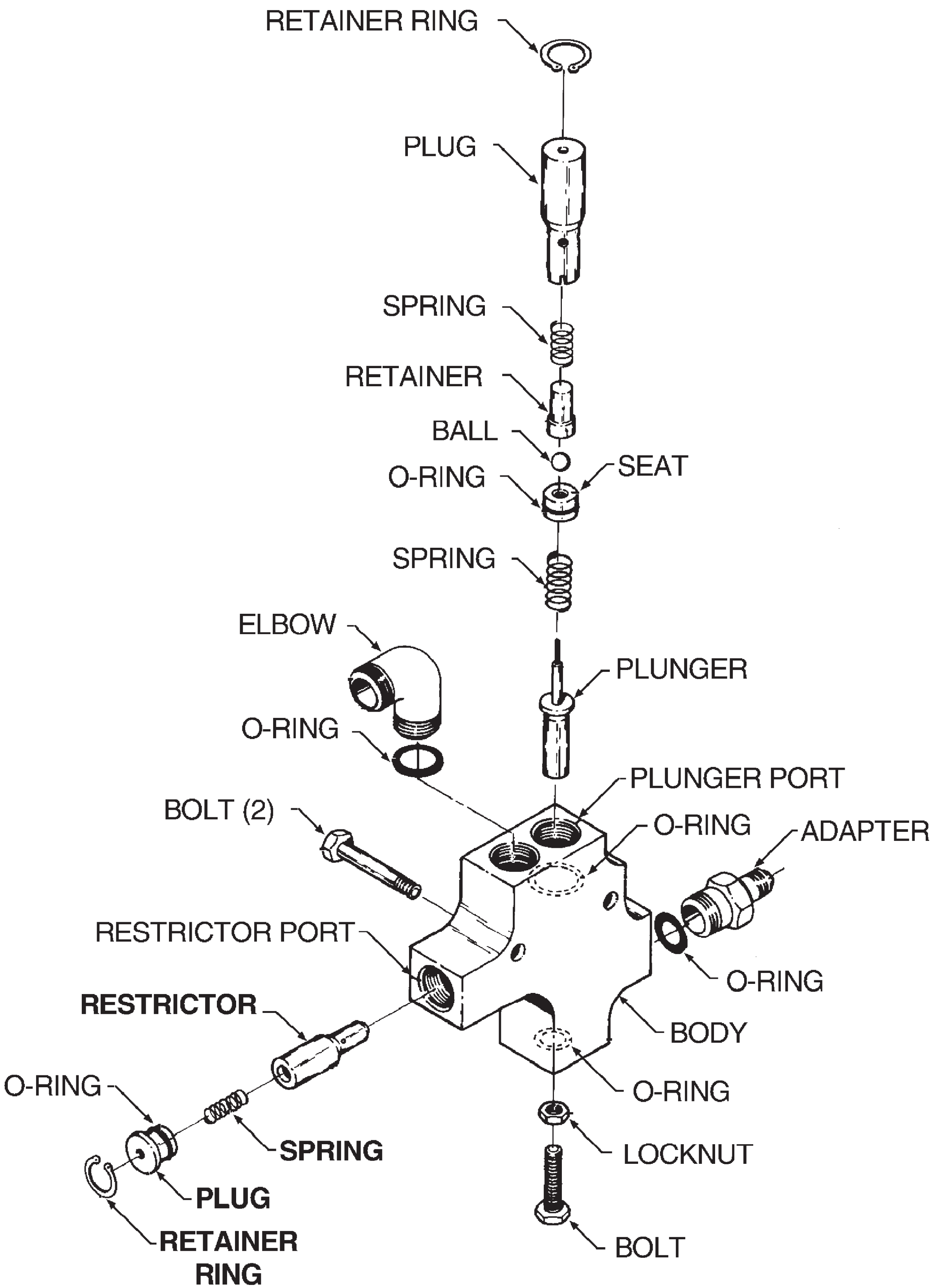
R. Thread locknut onto bolt and thread bolt into plunger. Torque locknut to 12 - 15 in.-lbs.

S. Purge sequence valve per “Component Purge Procedures” on page 323038, before installation.

T. Upon installation, set the sequence valve as described under Landing Gear Retraction System Functional Test, above.



Sequence Valve and Lines
Figure 11



Sequence Valve (Parker Hannifin)
Figure 12

8. Emergency Release Valve (Parker Hannifin)

Factory installed in S/N's 4636001–4636298, 4636300–4636313. See "Figure 13" on page 323035.

Emergency gear release valve is located beneath floor boards under copilot's seat. For access, remove seat, carpet, access panel, and center floorboard.

A. Removal and Disassembly

(1) Remove and quickly cap hydraulic lines.
(2) Loosen nuts at bulkhead fittings.
(3) Disconnect cable.
(4) Remove valve from aircraft.
(5) Drain valve by removing two caps.
(6) Remove lockwire, two bolts, and two washers that secure bracket. Remove bracket, lever, and links as one unit.
(7) Pull actuating rod out of end gland. Remove and discard O-ring from actuating rod.
(8) Remove end glands from UP and DN ports. Remove and discard O-rings from end glands.
(9) Remove bulkhead fittings from UP and DN ports. Remove and discard O-rings from bulkhead fittings.
(10) Remove check valves. Remove and discard O-ring from inside valve body.

B. Assembly and Installation

WARNING: USE SOLVENT IN WELL VENTILATED AREA. AVOID CONTACT WITH SKIN. KEEP AWAY FROM HEAT OR FLAME.

CAUTION: DO NOT REUSE LOCKWIRE, COTTER PINS, OR O-RINGS. COAT REPLACEMENT O-RINGS WITH HYDRAULIC FLUID MIL-PRF-5606.

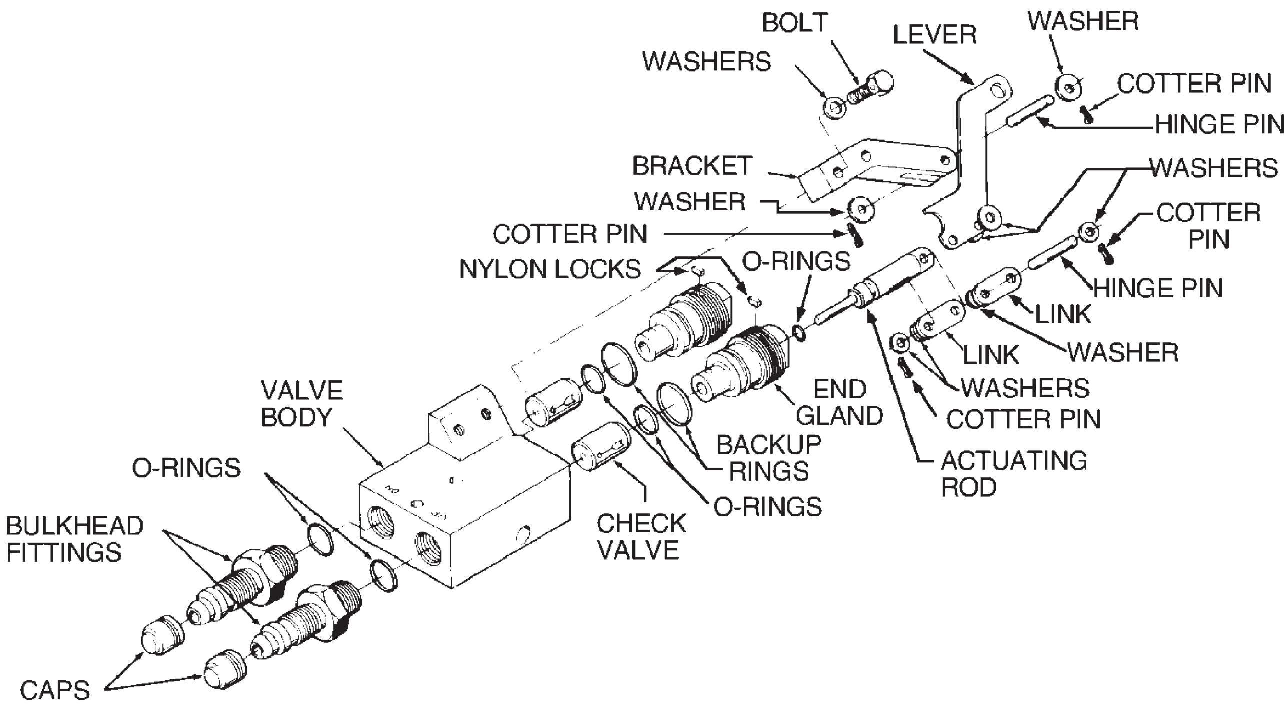
(1) Clean parts as needed with solvent. Wipe dry.
(2) Lubricate all parts in valve body with hydraulic fluid.
(3) Install new O-rings onto bulkhead fittings.
(4) Install bulkhead fittings into UP and DN ports. Torque to 60–65 in·lbs.

CAUTION: CHECK VALVES ARE NOT INTERCHANGEABLE. BEFORE INSERTING VALVES INTO UP AND DN PORTS, CORRECTLY IDENTIFY VALVE BY PRINTED VENDOR NUMBER.

(5) Insert check valve with rubber seal (2203C-20) into DN port. Arrow on check valve must point toward end gland. Push seal against valve.
(6) Install O-ring onto end gland assembly.
(7) Thread end gland assembly into DN port. Torque to 60–65 in·lbs.
(8) Insert check valve with teflon seal (2203C-3-5) into UP port. Arrow on check valve must point toward end gland. Push seal against valve.
(9) Install O-ring onto end gland assembly.
(10) Thread end gland assembly into UP port. Torque to 60–65 in·lbs.
(11) Attach actuating rod to bracket, lever, and links. lubricate actuating rod with hydraulic fluid.
(12) Install O-ring onto actuating rod. Insert rod into end gland assembly.
(13) Slide bracket/lever assembly as far as possible toward check with lever in open position.

NOTE: There should be very little travel.

(14) Torque bolts to 20–25 in·lbs. Lockwire bolts.



Emergency Release Valve (Parker Hannifin)
Figure 13

(15) Connect cable.

(16) Uncap and connect hydraulic lines.

(17) Install access panel and floorboard. Secure with screws. Replace carpet and seat.

9. Emergency Gear Release Valve (Frisby/Triumph)

Factory installed in S/N's 4636299, 4636314 and up, and 4692001 and up. See "Figure 14" on page 323037.

CAUTION: DO NOT PULL THE EMERGENCY GEAR RELEASE KNOB WHILE THE LANDING GEAR ARE DOWN AND LOCKED. DOING SO CAN DAMAGE THE EMERGENCY GEAR RELEASE VALVE.

The emergency gear release valve is located beneath floor boards, under co-pilot's seat. For access, remove seat, carpet, access panel, and center floorboard.

NOTE: No field repair of Triumph / Frisby emergency gear release valves installed in these airplanes is authorized. Emergency gear release valve overhaul service is available through Piper dealers.

A. Removal

(1) Place airplane on jacks per 7-10-00.
(2) Reduce hydraulic pressure as follows:
  (a) Place gear control in the UP position.
  (b) Allow gear to retract until pressure drops (approximately halfway).
  (c) Pull Hydraulic Pump circuit breaker.

  NOTE: Have assistants standing by each main gear to catch the gear and prevent them from falling back to a down and locked position. Block each main gear to hold that position.

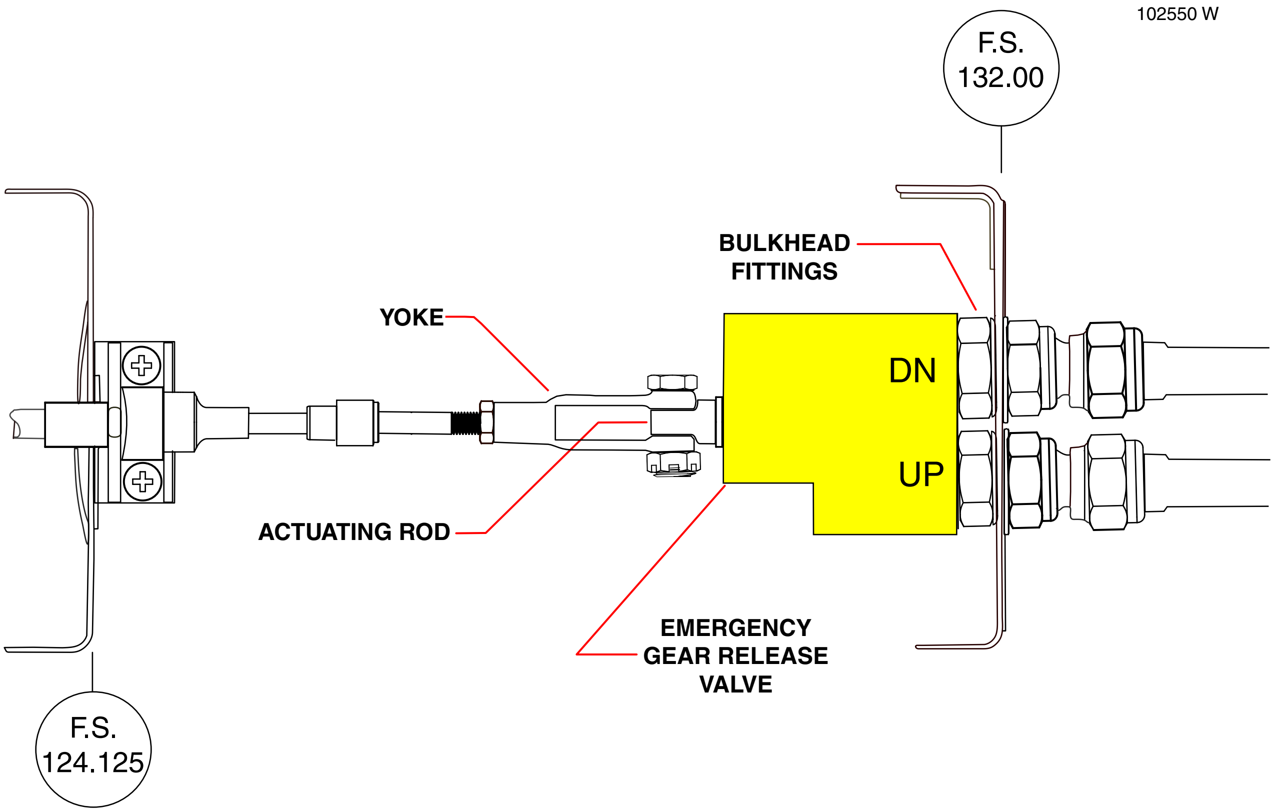
  (d) Set BATT MASTER switch to the OFF position.
(3) Disconnect and quickly cap hydraulic lines.
(4) Loosen and remove bulkhead fitting nuts from the aft side of the bulkhead.
(5) Disconnect yoke from actuating rod.
(6) Remove valve from aircraft.

B. Installation

(1) Attach actuating rod to yoke.
(2) Position valve at bulkhead and install and tighten bulkhead fitting nuts from aft side of bulkhead.
(3) Uncap and connect hydraulic lines. Purge system of air. (Refer to 29-10-00, Bleeding Hydraulic System.)
(4) Pull the hydraulic pump circuit breaker out.
(5) Release the guard and pull gear free fall knob out.
(6) Verify that the landing gear falls to the down and locked position.

NOTE: The main gear may not fall completely into the locked position during ground free fall test. Extend the gear using a side force at the center line of axle that does not exceed nine (9) pounds. The gear should then lock.

(7) Place the gear selector switch in the DOWN position.
(8) Push in the free fall knob. Position the guard.
(9) Push the hydraulic pump circuit breaker in.
(10) Cycle gear up and down, one time to verify normal operations.
(11) Check that hydraulic pump circuit breaker is set (pressed in), gear control switch is in the DOWN position and BATT MASTER switch is OFF.
(12) Remove airplane from jacks. (Refer to 7-10-00).
(13) Install access panel and floorboard. Secure with screws. Replace carpet and seat.



Emergency Gear Release Valve
Figure 14

Effectivity
4636314 & up, and
4636314 thru 4636347 with
Kit No. 767-364 installed,
and 4692001 and up

10. Component Purge Procedures

(PIR-PPS60175-2, Rev. B.)

WARNING: ONCE EACH OF THE FOLLOWING STEPS IN THIS SECTION HAS BEEN PERFORMED, THE SYSTEM MUST NOT BE SUBJECT TO AIR CONTAMINATION AGAIN. IF IT IS NECESSARY TO CHANGE THE ACTUATORS POSITION, THIS MUST BE DONE VIA HYDRAULIC POWER SUPPLY PUMP, MANUAL, OR ELECTRIC, USING THE PROPER FLUID AND TAKING CARE NOT TO INTRODUCE AIR BACK INTO THE SYSTEM. IN THE EVENT AIR IS INTRODUCED, THE APPLICABLE FILL AND PURGE PROCEDURE(S) MUST BE REPEATED.

NOTE: A 28 volt (direct current) regulated power source capable of connection to the external power receptacle of the aircraft and of supplying a minimum of 50 amperes and adequate supply of clean MIL-PRF-5606 Hydraulic Fluid, Hydraulic cart, and clean hydraulic fittings is required to accomplish this procedure.

A. Fill each two-port actuator or actuator assembly (nose gear, main gear) by performing the following steps prior to its installation on the aircraft:

(1) Attach a hydraulic power supply to the hoses or fittings of the actuator assembly.

(2) Orient the actuator such that the body is horizontal and rotate it such that the fittings are at the 12 o'clock position.

(3) Using the hydraulic power supply, extend and retract the rod until all air is removed (no foam or bubbles).

(4) Using the hydraulic power supply, extend the rod.

(5) Remove the hydraulic power supply and plug the actuator assembly hoses with AN806 plugs or AN929 caps as appropriate while ensuring air is not reintroduced into the system.

(6) Put a protective sleeve over the rod end.

NOTE: Do not exceed 100 psi.

B. Fill each one-port actuator (nose gear door) by performing the following steps prior to its installation on the aircraft:

(1) Manually retract the rod fully.

(2) Bleed the hose with an MS51527 fitting and install it into the actuator port.

(3) Dip the hose into a reservoir of hydraulic fluid and slowly extend the rod to fill the actuator.

(4) Remove the hose and fitting and plug actuator with an AN814 plug while ensuring air is not reintroduced into the system.

(5) Put a protective sleeve over the rod end.

C. Nose Landing Gear (NLG) Sequence Valve & Lines

See "Figure 11" on page 323032.

(1) Connect the power supply hose (A) to the system Sequence Valve hose (C).

(2) Connect the other power supply hose (B) to the Door Actuator hose (D).

(3) Depress sequence valve plunger (F) and run the hydraulic power supply to flush from hose (A) to hose (B) until air is removed (no foam or bubbles).

(4) Remove power supply hoses (A, B) and put AN806 plugs in the hoses (C, D) while ensuring air is not reintroduced into the system.

WHEELS AND BRAKES

1. Main Wheel Assembly

See "Figure 1".

A. Removal and Disassembly

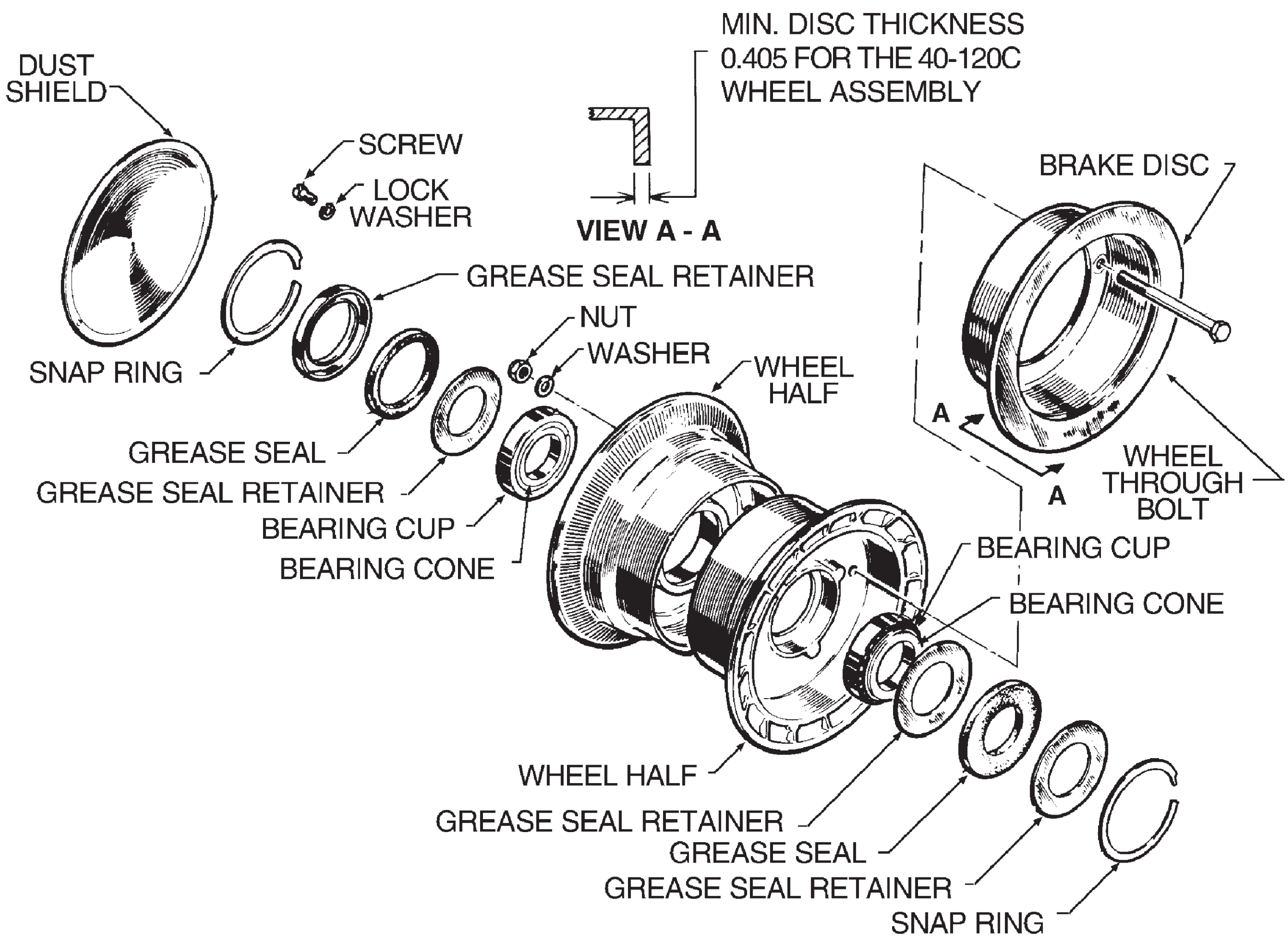
(1) Place the airplane on jacks per Jacking, 7-10-00.

(2) To remove the main wheel, remove the cap bolts that join the brake cylinder housing and the lining back plate assemblies. Remove the back plate from between brake disc and wheel.

(3) Remove the dust cover, the cotter pin and flat head pin that safeties the wheel nut, and the wheel nut. Slide the wheel from the axle.

(4) The wheel halves may be separated by first deflating the tire. With the tire sufficiently deflated, remove the wheel through bolts. Pull the wheel halves from the tire by removing the inner half from the tire first, and then the outer half.

(5) The wheel bearing assemblies may be removed from each wheel half by first removing the retainer snap rings that secure the grease seal retainers, and then the retainers, grease seals and bearing cone. The bearing cups should not be removed except for replacement. Refer to Nose and Main Wheel Assembly Repairs, below, for bearing cup replacement instructions.



Main Wheel Assembly
Figure 1

B. Assembly and Installation

CAUTION: DO NOT USE POWER TOOLS TO INSTALL NUTS AND BOLTS.

NOTE: Piper recommends that all tires on an airplane be from the same manufacturer and that the two main tires be the same part number.

NOTE: Tubes used must be from the same manufacturer as the tire. Do not mix tube and tire brands.

(1) Wheel Assembly

(a) Check that bearing cup for each wheel half is properly installed.

(b) Install tire with tube on wheel half with valve stem hole, inserting the stem through the valve hole. Align wheel bolt holes and then join two wheel halves.

(c) Install wheel through-bolts, washers and nuts to valve stem side, then tighten (draw up) the bolts in a criss-cross pattern to about 20 inch-pounds.

NOTE: Both plain and countersunk washers may be used. Some designs use bolts that have a radius between the head and shank, which requires the countersunk washers. Inspect bolts and washers prior to assembly. Properly oriented washers with countersunk surfaces to sit flush against the bolt head. Install washers and nuts on bolts (bolt heads are to be on brake disc side of wheel).

(d) Remove all pressure in the tire.

(e) Torque each nut again in a criss-cross pattern to about 45 inch-pounds before setting the final torque specified on the wheel placard. Do not use power tools to torque nuts.

(f) Inflate the tire to the specified pressure per Chart 1, 6-00-00.

(2) Bearing Installation

CAUTION: DO NOT MIX AVIATION WHEEL BEARING GREASES WITH EACH OTHER.

NOTE: Mobil Aviation Grease SHC 100 has been used by the wheel manufacturer since March 2007 and is the preferred grease. If not available, greases meeting MIL-PRF-81322 or DOD-G-24508A are suitable alternatives. If old grease is unknown, thoroughly clean bearing components before packing with new grease.

CAUTION: HANDLE BEARING CONES WITH EXTREME CARE TO PREVENT CONTAMINATION OR DAMAGE.

(a) Pack the bearing cones as follows:

NOTE: Pack the bearing cones just before installation to prevent contamination.

NOTE: Bearing cones can be packed by hand or by using a mechanical bearing greaser. The mechanical bearing greaser will do a more thorough job of packing the grease.

1) Clean the bearing cones.

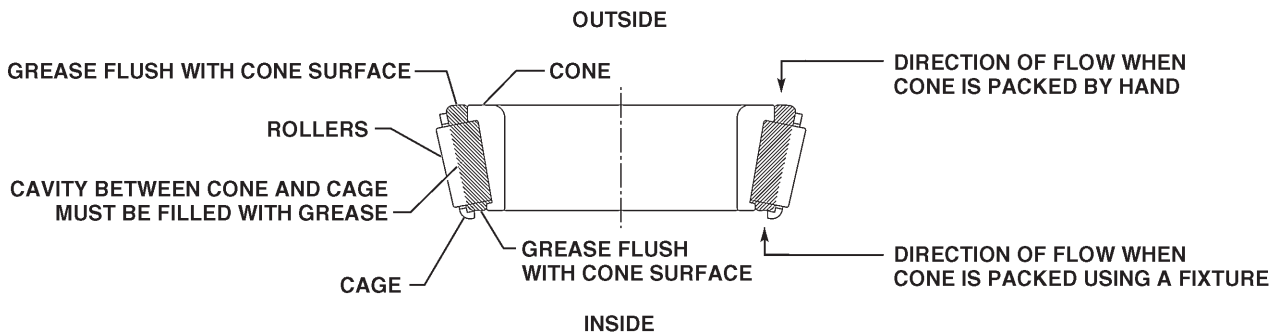
2) Push and force the grease up and out between the rollers, cone and cage.

3) The bearing is properly greased when no voids or daylight can be observed between the rollers and inner and outer races.

4) Disperse excess grease around each end and the tapered sides of each cone.

NOTE: Shaded area in “Figure 2” on page 32403 shows the recommended quantity of grease.

(b) Liberally swab the bearing cup, bearing bore hub and grease seal/snap ring areas with bearing grease.



Greasing Wheel Bearing
Figure 2

(c) If felt seals are used, lightly coat all surfaces of the felt with the wheel bearing grease. If rubber lip seals are used, lightly coat the rubber surfaces with bearing grease.

(d) Install the bearing cones, grease seals (felts and rings or rubber lip seals) and snap rings.

Excess grease will squeeze out. Remove the excess grease with an inward rotating movement against the bearing cone ID. Disperse any small amounts of grease on the exterior surface of the grease seal and snap ring and remove any grease from the hub outside surface.

(e) Use care that bearing grease does not become contaminated.

(3) Slide the wheel on the axle and secure with the retainer nut. While rotating the wheel, tighten the axle nut from 17 to 25 inch-pounds torque. Back off the nut to reduce the torque to zero. While rotating the wheel, tighten the axle nut from 10 to 15 inch-pounds torque. Rotate the axle nut to align the nearest slot with the cotter pin hole. Install the cotter pin.

(4) Position the brake lining back plates between the wheel and brake disc and the brake cylinder on the torque plate. Insert the spacer blocks between the back plates and cylinder, and install the four bolts to secure the assembly. If the brake was disconnected, reconnect the line and bleed the brakes.

C. Inspection

(1) Visually check all parts for cracks, distortion, defects and excess wear.

(2) Check tie bolts for looseness or failure.

(3) Check internal diameter of felt grease seals. Replace the felt grease seal if surface is hard or gritty.

(4) Check tire for cuts, internal bruises and deterioration.

(5) Check bearing cones and cups for wear and pitting. Then lubricate.

(6) Replace any wheel casting having visible cracks.

(7) Inspect brake disc for cracks, excessive wear or scoring, rust, corrosion and warpage.

(8) Remove rust and blend out nicks, using fine 400 grit sandpaper.

(9) Replace disc if cracked or when disc is worn below .405 in. (Wheel Brake Assembly - Cleaning, Inspection and Repair, below.)

1. Nose Wheel Assembly

See “Figure 3” on page 32405.

A. Removal and Disassembly

(1) Jack the airplane enough to raise the nose wheel clear of the ground (per Jacking, 7-10-00).

(2) To remove the nose wheel, first remove the cotter pin and washer that secures the safety clevis pin of the wheel nut. Next remove the clevis pin and wheel nut. Then slide the wheel from the axle.

(3) The wheel halves may be separated by first deflating the tire. With the tire sufficiently deflated, remove the wheel through bolts. Pull the wheel halves from the tire by removing the wheel half opposite the valve stem first and then removing the other half.

(4) The wheel bearing assemblies may be removed from each wheel half by first removing the snap rings that secure the grease seal retainers, and then the retainers, grease seals and bearing cones. The bearing cups should be removed by tapping out evenly from the inside.

B. Assembly and Installation

(PIR-PPS50025, Rev. AF.)

CAUTION: DO NOT USE POWER TOOLS TO INSTALL NUTS AND BOLTS.

NOTE: Piper recommends that all tires on an airplane be from the same manufacturer.

NOTE: Tubes used must be from the same manufacturer as the tire. Do not mix tube and tire brands.

(1) Check that bearing cup for each wheel half is properly installed.

(2) Install tire with tube on wheel half with valve stem hole, inserting the stem through the valve hole. Align wheel bolt holes and then join two wheel halves.

(3) Install wheel through-bolts, washers and nuts to valve stem side, then tighten (draw up) the bolts in a criss-cross pattern to about 20 inch-pounds.

NOTE: Both plain and countersunk washers may be used. Some designs use bolts that have a radius between the head and shank, which requires the countersunk washers. Inspect bolts and washers prior to assembly. Properly oriented washers with countersunk surfaces to sit flush against the bolt head. Install washers and nuts on bolts (bolt heads are to be on brake disc side of wheel).

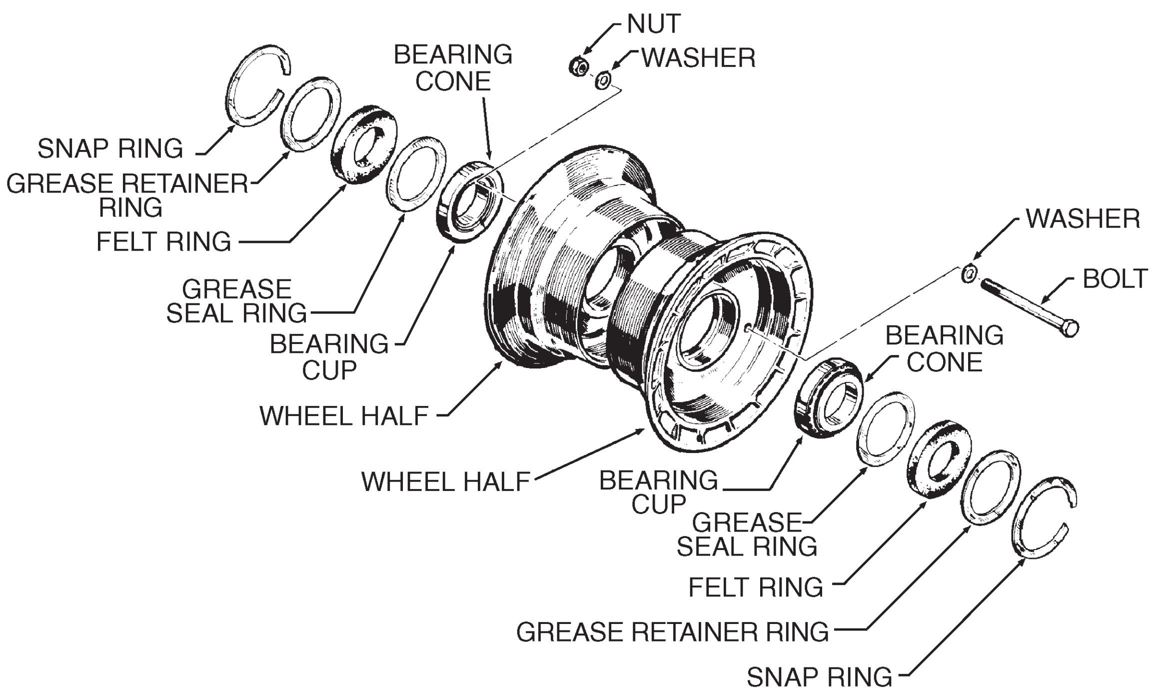
(4) Remove all pressure in the tire.

(5) Torque each nut again in a criss-cross pattern to about 45 inch-pounds before setting the final torque specified on the wheel placard. Do not use power tools to torque nuts.

(6) Inflate the tire to the specified pressure per Chart 1, 6-00-00.

(7) Grease and install wheel bearings per Bearing Installation under Main Wheel Assembly, Assembly and Installation, below.

(8) Slide the wheel on the axle. While rotating the wheel, tighten the axle nut from 17 to 25 inch-pounds. Back off the nut to reduce torque to zero. While rotating the wheel, tighten the axle nut from 10 to 15 inch-pounds. Rotate the axle nut to align the nearest slot with the cotter pin hole. Install the cotter pin.



Nose Wheel Assembly
Figure 3

C. Inspection

(1) Visually check all parts for cracks, distortion, defects and excess wear.

(2) Check tie bolts for looseness or failure.

(3) Check internal diameter of felt grease seals. Replace the felt grease seal if surface is hard or gritty.

(4) Check tire for cuts, internal bruises and deterioration.

(5) Check bearing cones and cups for wear and pitting. Then lubricate.

(6) Replace any wheel casting having visible cracks.

2. Nose and Main Wheel Assembly Repairs

Repairs are limited to blending out small nicks, scratches, gouges and areas of slight corrosion, plus the replacement of parts which are cracked or badly corroded.

NOTE: Remove corrosion and blend out small nicks, using fine 400 grit sandpaper.

Wheels may also be repainted if the parts have been repaired and thoroughly cleaned. Paint exposed areas with one coat fluid resistant epoxy primer and one coat of aluminum lacquer.

NOTE: Never paint working surfaces of the bearing cups.

3. Bearing Cup

A. Removal

(1) Insert wheel half into boiling water for 15 minutes or place in an oven not exceeding 250° F (121° C) for 15 minutes.

(2) Remove from source of heat and invert wheel half. If the cup does not drop out, tap the cup evenly from the axle bore with a fiber drift pin or suitable arbor press.

B. Installation

WARNING: USE PROTECTIVE GLOVES WHEN WORKING WITH DRY ICE OR HANDLING HEATED PARTS.

(1) Place wheel half in an oven not exceeding 212°F (100°C) for 15 minutes. Chill new bearing cup in an atmosphere of -25°F to -65°F for no less than 4 hours. Chilling may also be accomplished by placing the bearing cup in dry ice for a minimum of 15 minutes.

(2) To install a new bearing cup, apply one coat of zinc chromate to cup bore.

(3) Remove wheel half from heat source and remove bearing cup from cold source. Dry cup thoroughly.

(4) Install the chilled bearing cup into bearing bore of heated wheel half. Tap gently into place with a fiber drift making sure cup is evenly seated against shoulder of wheel half. Avoid cocking bearing cup during installation. If bearing cup will not seat properly in wheel half, repeat above procedure or replace wheel half assembly.

4. Brake Adjustment and Lining Tolerance

No adjustment of the brake lining clearance is necessary, as the brakes are self-adjusting. Inspection of the lining is necessary, and it may be inspected visually while installed on the airplane. The linings are of the riveted type and should be replaced if the thickness of any one segment becomes worn below 0.100 of an inch or unevenly worn.

5. Wheel Brake Assembly

See “Figure 4” on page 32407 and “Figure 5” on page 32408.

A. Removal and Disassembly

To remove the brake assembly:

(1) Disconnect the brake line from the brake cylinder at the tube fitting.

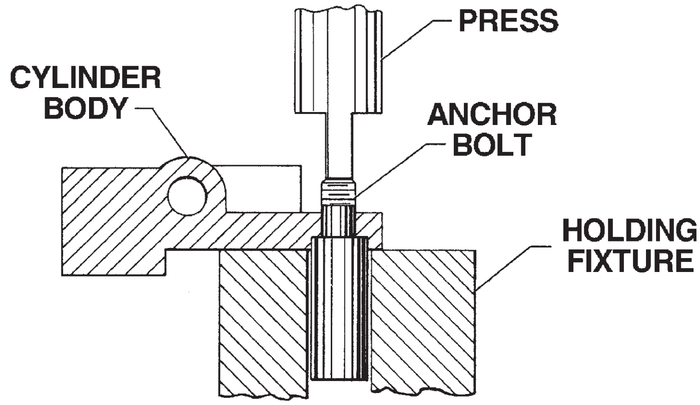
(2) Remove the cap bolts that join the brake cylinder housing and the lining back plate assembly. Remove the back plate from between the brake disc and wheel.

(3) Slide the brake cylinder housing from the torque plate.

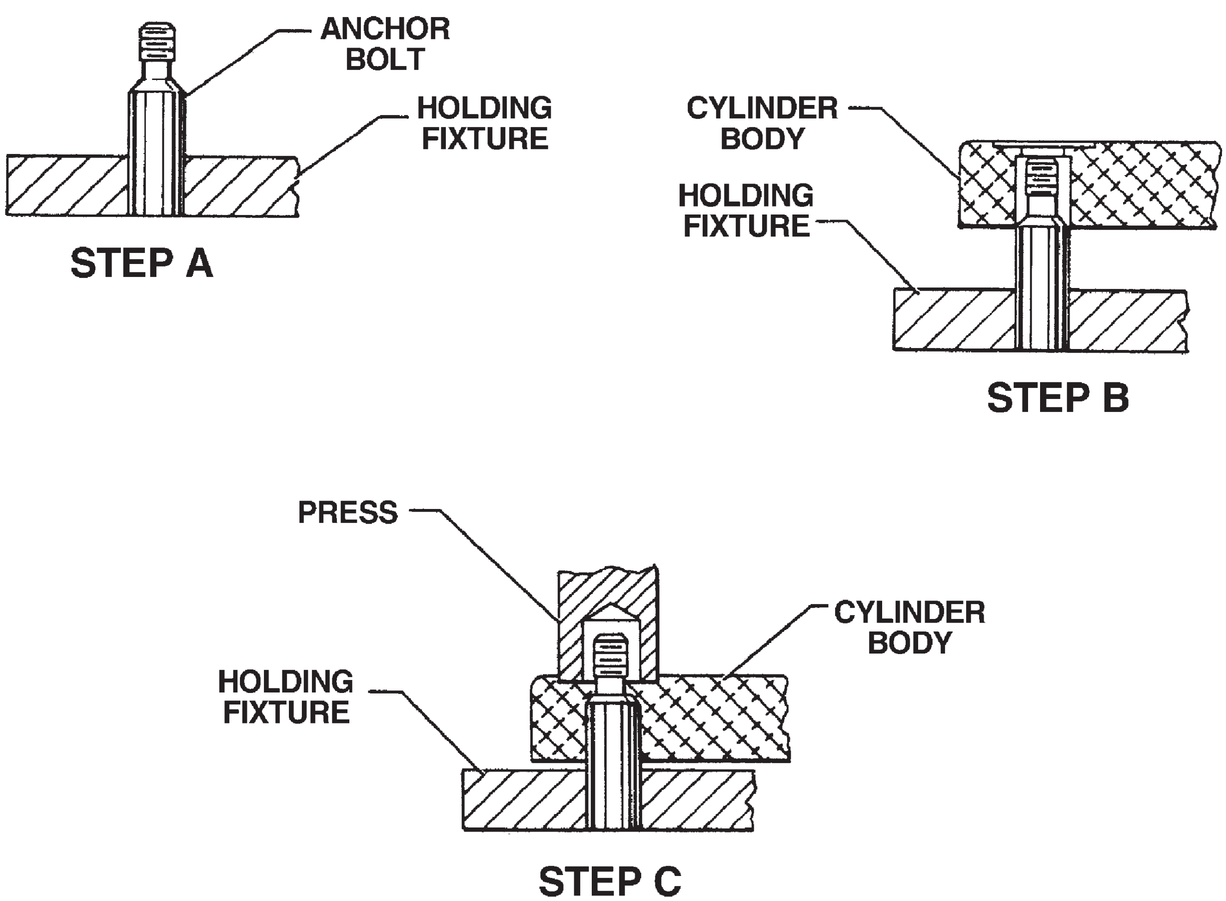
(4) Remove the pressure plate by sliding it off the anchor bolts of the housing.

(5) Remove piston(s) by injecting low air pressure in the cylinder fluid inlet and forcing the piston from the housing.

REMOVAL



INSTALLATION



Anchor Bolt, Removal and Installation
Figure 4

(6) Check anchor bolt for wear.

(a) To remove anchor bolt:

1) Position cylinder assembly on a holding fixture (refer to “Figure 4”).

2) Use a suitable arbor press to remove the anchor bolt from the cylinder body.

(b) To install anchor bolt:

1) Support anchor bolt in a holding fixture (refer to “Figure 4”, Step A).

2) Align cylinder body over anchor bolt (refer to “Figure 4”, Step B).

3) Use a suitable arbor press and apply pressure on the spot face directly over the anchor bolt hole (refer to “Figure 4”, Step C).

B. Assembly and Installation

(1) Lubricate the piston O-ring(s) with fluid MIL-PRF-5606 and install on piston(s). Slide the piston(s) in cylinder housing until flush with surface of housing.

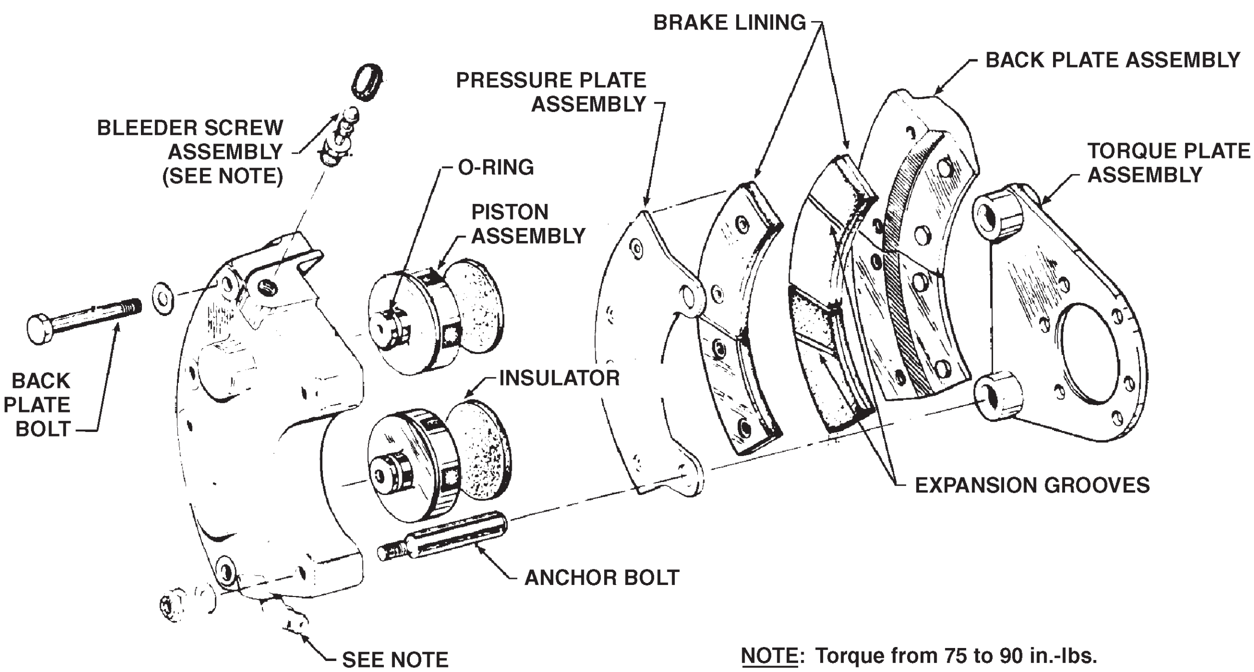
(2) Slide the lining pressure plate onto the anchor bolts of the housing.

(3) Slide the cylinder housing assembly on the torque plate of the gear.

(4) Position the lining back plate between the wheel and brake disc. Install the bolts and dry torque to 90 inch-pounds to secure the assembly.

(5) Connect the brake line to the brake cylinder housing.

(6) Bleed the brake system as described in Bleeding Brakes.



Wheel Brake Assembly
Figure 5

C. Cleaning, Inspection, and Repair

See “Figure 5” on page 32408.

(1) Clean the assembly with a suitable solvent and dry thoroughly.

(2) Check the wall of the cylinder housing and piston for scratches, burrs, corrosion, etc, that may damage O-rings.

(3) Check the general condition of the brake bleeder screw and lines.

(4) Check the brake disc for wear, grooves, scratches, or pits. The minimum disc thickness of Disc 164-46 used on Wheel Assembly 40-120C is .405. A single groove or isolated grooves up to .030 of an inch deep would not necessitate replacement, but a grooving of the entire surface would reduce lining life and would necessitate replacement of the disc. Should it be necessary to remove the wheel disc, see Main Wheel Assembly - Removal and Disassembly.

(5) To remove the snap-on type lining used on the brake assemblies, pry it loose with a screwdriver or a thin flat wedge. Install the snap-on type lining by positioning it onto the pins and applying pressure to snap into position.

(6) If a powdery rust appears on the brake disc, one or two taxi braking applications should wipe the disc clear. Rust allowed to progress beyond this point may require removal of the disc from the wheel assembly to properly clean both faces. Wire brushing followed by sanding with 220 grit sandpaper can restore the braking surfaces for continued use.

NOTE: To inspect the linings, check the expansion groove. If the groove is not showing, replace the lining.

D. Lining Conditioning Procedures

When new linings have been installed, it is important to condition them properly to obtain the service life designed into them. The metallic and organic linings are not conditioned in the same manner because they have different operating characteristics. Separate conditioning procedures are given for metallic and organic linings.

(1) Nonasbestos Organic Linings

(a) Taxi aircraft for 1500 feet with engine at 1700 rpm applying brake pedal force as needed to develop a 5–10 mph taxi speed.

(b) Allow the brakes to cool for 10 to 15 minutes.

(c) Apply brakes and check for restraint at high static throttle. If brakes hold, conditioning is complete.

(d) If brakes cannot hold aircraft during static run up, allow brakes to completely cool, and repeat steps (a) through (c).

(2) Metallic Linings

(a) Perform two (2) consecutive full stop braking applications from 30–35 knots. Do not allow the brake discs to cool substantially between the stops.

(b) Allow the brakes to cool for 10-15 minutes.

(c) Apply brakes and check for restraint at high static throttle. If brakes hold, conditioning is complete.

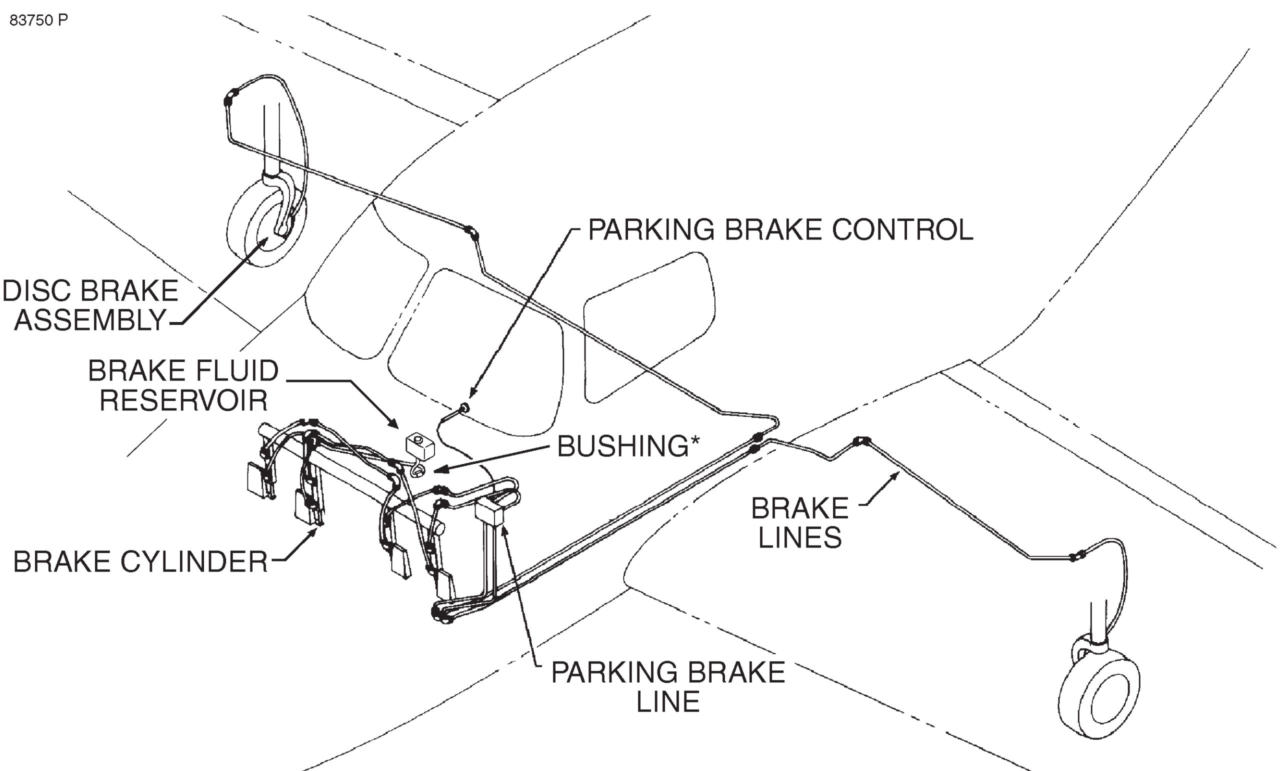
(d) If brakes cannot hold aircraft during static runup, allow brakes to completely cool, and repeat steps (a) through (c).

This conditioning procedure will wear off high spots and generate sufficient heat to create a thin layer of glazed material at the lining friction surface. Normal brake usage should generate enough heat to maintain the glaze throughout the life of the lining.

Properly conditioned linings will provide many hours of maintenance free service. A visual inspection of the brake disc will indicate the lining condition. A smooth surface, one without grooves, indicates the linings are properly glazed. If the disc is rough (grooved), the linings must be reglazed. The conditioning procedure should be performed whenever the rough disc condition is observed. Light use, such as in taxiing, will cause the glaze to be worn rapidly.

6. Brake System Installation

See “Figure 8”.



* Final torque bushing to 80 IN/LBS ± 10 IN/LBS.

NOTE: Apply Tite Seal 3-T3 to all male pipe threads prior to assembly. Do not allow sealant to enter system.

Brake System Installation
Figure 6

7. Toe Brake Cylinder Assembly

See “Figure 7” on page 324012.

A. Removal

WARNING: MAKE CERTAIN AIRPLANE IS PARKED ON LEVEL GROUND WITH WHEELS CHOCKED.

CAUTION: USE PARKER HANNIFIN REPLACEMENT BRAKE CYLINDERS ONLY. GAR-KENYON AND PARKER HANNIFIN BRAKE CYLINDERS ARE NOT INTERCHANGEABLE. PILOT'S LEFT AND RIGHT CYLINDERS MUST BE THE SAME MANUFACTURER. COPILOT'S LEFT AND RIGHT CYLINDERS MUST BE THE SAME MANUFACTURER. SEE PIPER PARTS CATALOG, P/N 761-878.

(1) Place chocks under main and nose wheels.

(2) Release parking brake.

(3) Disconnect and quickly cap upper and lower hydraulic lines from cylinder assembly being removed.

(4) Disconnect clevis from rudder pedal by removing cotter pin and pin.

NOTE: The rod and ring subassembly may be removed at this time without further disassembly of the toe brake assembly. Refer to Disassembly, Repair, and Installation, below, steps (1) and (2) for rod and ring removal instructions and steps (9) and (10) for installation. (See also the CAUTION above step (9).)

(5) (Inboard toe brake assemblies only.) Remove nut, bolt, and washer holding nose wheel steering rod assembly to pedal.

(6) Remove the two nuts, bolts, and washers holding cylinder assembly to rudder bar.

(7) Remove toe brake and cylinder assembly from airplane.

B. Disassembly, Repair, and Installation

(1) Remove end fitting from cylinder housing.

(2) Slide entire rod and ring assembly with attached piston out of cylinder housing.

NOTE: Repairs to the rod and ring assembly and piston are limited to polishing small scratches and burrs, and replacement of parts.

(3) Clean cylinder bore, rod and ring assembly, and piston with dry-type solvent.

(4) To replace any of the component parts shown in the parts catalog, remove retaining ring. All parts will slide off rod and ring assembly.

(5) Soak end fitting O-rings (one internal, one external) and piston O-ring in hydraulic fluid. Coat with fluid-compatible gel.

(6) Install power spring on rod and ring assembly.

(7) Insert rod and ring sub-assembly with spring through end fitting.

(8) Insert, in order: washer seal, piston, piston spring, and bushing. Snap retaining ring into groove.

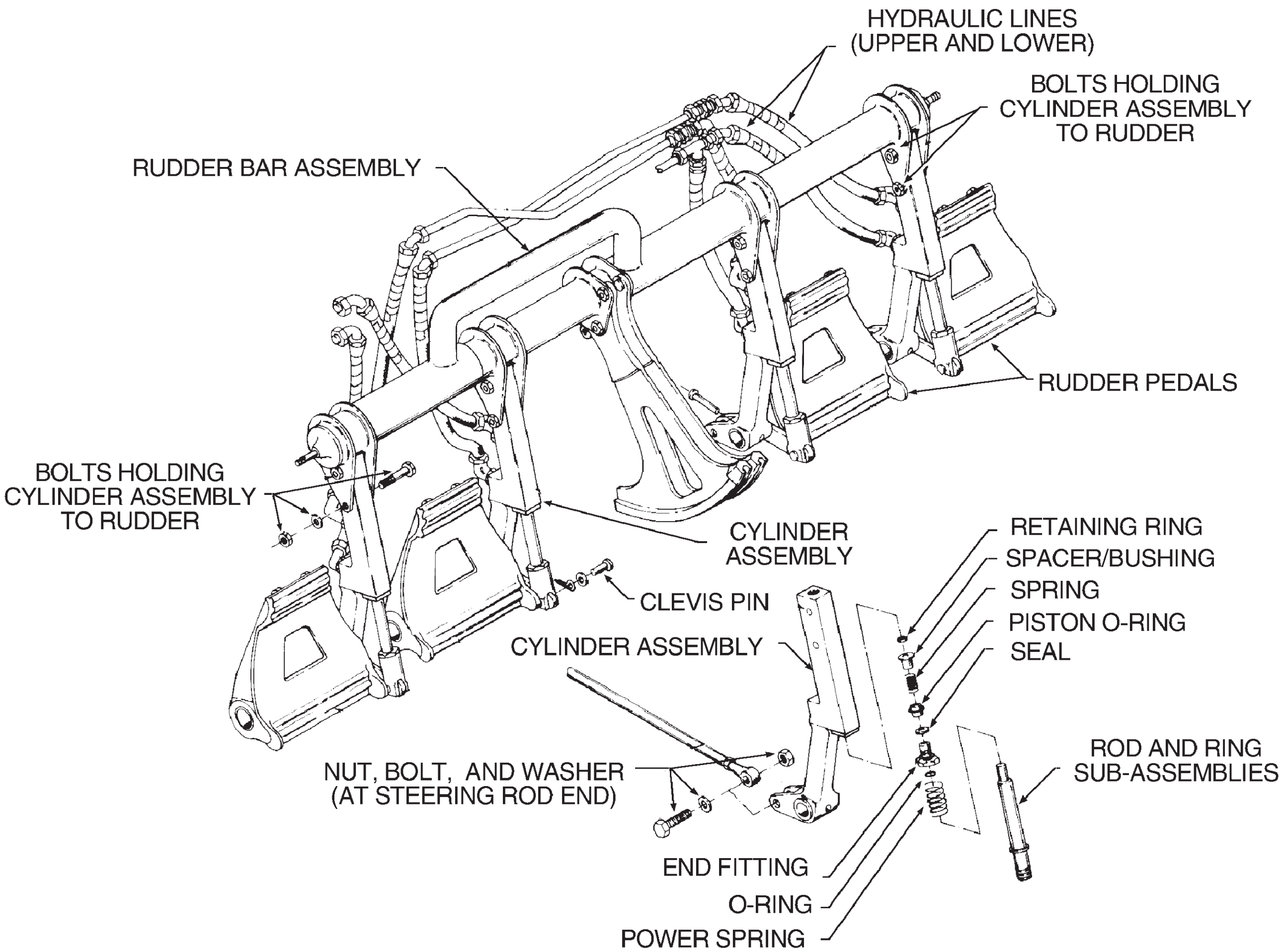
CAUTION: PISTON MUST BE INSERTED INTO CYLINDER HOUSING IN SUCH A WAY THAT ONLY PISTON O-RING CONTACTS THE CYLINDER INNER SURFACE. OTHERWISE, THE INNER SURFACE MAY BECOME SCRATCHED.

(9) Carefully insert piston and rod and ring assembly into cylinder housing.

(10) Tighten end fitting to a torque of 13-17 inch-lbs.

(11) Connect cylinder assembly to rudder bar with two nuts, bolts, and washers.

(12) (Inboard toe brake assemblies only.) Connect pedal to nose wheel steering rod assembly.



Toe Brake Cylinder Assembly
Figure 7

(13) Connect clevis to rudder pedal by inserting pin, washer, and cotter pin.

(14) Uncap and connect upper and lower hydraulic lines to cylinder assembly.

(15) Bleed brakes (refer to “Brake Bleeding Procedure” on page 324013).

(16) Set parking brakes.

(17) Remove chocks from wheels before moving airplane.

8. Brake Bleeding Procedure

A. Setup

(1) Place rudder pedals in neutral position and ensure that all four master cylinders are in full up position.

(2) Fully release parking brake (i.e., parking brake valve open).

B. Gravity

(1) On both main landing gear wheel brake assemblies, attach a clear plastic hose to the brake bleeders. Extend the hoses into a container partially filled with hydraulic fluid, MIL-PRF-5606. The ends of the hoses should be submerged in the fluid.

(2) Fill the brake reservoir on the forward pressure bulkhead with hydraulic fluid, MIL-PRF-5606.

NOTE: During the procedure, have someone monitor the fluid level and ensure the brake reservoir is kept topped up. Failure to do so will allow air into the system and require the procedure be restarted.

(3) Depress co-pilot left brake to maximum and hold.

(4) Open the left brake bleeder approximately one and one-half to two turns to bleed out air and fluid until brake pedal almost bottoms out.

(5) Immediately close left brake bleeder. Release brake pedal to neutral.

(6) Repeat steps “(3)”, “(4)”, and “(5)” until all air bubbles have escaped from the left brake bleed port.

(7) Repeat steps “(3)”, “(4)”, “(5)”, and “(6)” with pilot’s left brake.

(8) Shift to the right brake bleeder and repeat steps “(3)”, “(4)”, “(5)”, and “(6)” with with co-pilot’s right brake.

(9) Repeat steps “(3)”, “(4)”, “(5)”, and “(6)” with pilot’s right brake.

(10) Tighten both wheel bleeders.

(11) Check per “Brake System Leak Check” on page 324014.

C. Pressure

The following procedure is recommended when system has been entirely drained of fluid and must be refilled.

(1) Place a small clear plastic hose on the vent tube of the brake reservoir and place a second small clear plastic hose on the bleeder fitting on one main landing gear. Place the open ends of these hoses in a suitable container to collect the fluid overflow. Open the bleeder fitting one or two turns.

(2) On the other main gear, slide the hose of the brake fluid pressure fill unit over the bleeder fitting, then open the fitting one or two turns and pressure fill the brake system with MIL-PRF-5606 fluid.

(3) With fluid continually flowing through the brake system, SLOWLY and together actuate the pedals several times.

NOTE: By watching the fluid pass through the plastic hose at the fluid reservoir and the bleeder fitting on the gear being bled, it can be determined whether any air is left in the system. If air bubbles are evident, filling of the system shall be continued until all the air is out of the system and a steady flow of fluid is obtained. Should the toe brakes remain spongy, it may be necessary to repeat the brake bleeding procedure. Or use the Gravity procedure, above.

(4) Close the open bleeder fitting on the gear being bled. Close the open bleeder fitting to which the pressure hose is attached; then close the pressure unit and remove the hoses from the bleeder fittings and from the reservoir vent. Check the brakes for proper pedal pressure. Replace the caps over the bleeder fittings.

NOTE: It may be necessary to remove any trapped air in the top of the wheel brake unit by applying pressure to the system with the toe brakes and slowly opening the bleeder and releasing the hand lever. Or, using the Gravity procedure, above.

(5) Repeat this procedure, if necessary, on the other gear.

(6) Drain excess fluid from the reservoir to fluid level line with a syringe.

D. Brake Bleeding After a Unit Has Been Changed

CAUTION: DO NOT ALLOW PRESSURE TO BLEED OFF BEFORE CLOSING BLEEDERS, FOR THIS WILL ALLOW AIR TO ENTER THE SYSTEM. REPEAT THE PUMPING AND BLEEDING APPROXIMATELY 10 OR MORE TIMES OR UNTIL ALL THE AIR IS RELEASED FROM THE SYSTEM. DURING ALL BLEEDING, FLUID LEVEL OF THE RESERVOIR MUST BE MAINTAINED.

(1) Actuate the toe brakes until some pressure builds up in the system. At this time crack the attaching B nuts at any of the hose connections of the replaced unit. Most of the brake pedals' spongy feeling should be displaced by this action. Tighten B nut.

(2) Actuate the toe brake/ master cylinder on the side on which the unit has been replaced. Bleed fluid through the wheel brake assembly by pumping pressure and cracking bleeder until the pressure drops.

9. Brake System Leak Check

A. Push on the toe brakes for a good firm pressure and lock the parking brake mechanism.

B. Allow the system to stand for approximately 10 minutes.

C. Visually inspect the entire system lines, fitting, and seals for deformation and external leakage. No visible leakage is allowed.

D. Then, when pushing the toe brakes, they should not be able to be pushed forward further than the original set.

E. If the toe brakes can be pushed forward and feel spongy, a leak is present at some point in the system.

F. This leak may appear at any one of the connections throughout the system or internally in the toe brake master cylinder or wheel brake assemblies.

10. Parking Brake Valve

See “Figure 8” on page 323025.

A. Removal

(1) Disconnect the parking brake cable from the valve actuating arm.

(2) Disconnect the fluid lines from the valve.

(3) Remove the screws that attach the valve to its mounting bracket.

(4) Place a protective material over the line openings to prevent contamination of the system.

B. Installation

(1) Attach the valve to the bulkhead mounting bracket with screws.

(2) Connect the fluid lines to the valve.

(3) Connect the control cable to the valve lever.

C. Disassembly

(1) Remove the two fittings from the outside of the valve body. A valve spring is held in place by the fittings. Use caution not to loosen these when removing the fittings.

(2) From the valve body, remove the valve spring and valve.

(3) To remove the valve cam, remove the nut, washer, bushing and spring. Pull the cam from the valve body.

D. Assembly

(1) Install O-rings on valve cam.

(2) Lubricate O-rings with fluid (MIL-PRF-5606), insert cam into valve body and secure with spring, bushings, washer and self-locking nut.

(3) Install O-ring on the valve, insert valve in hole of out port, install valve spring and secure with outlet fitting.

E. Cleaning, Inspection, and Repair

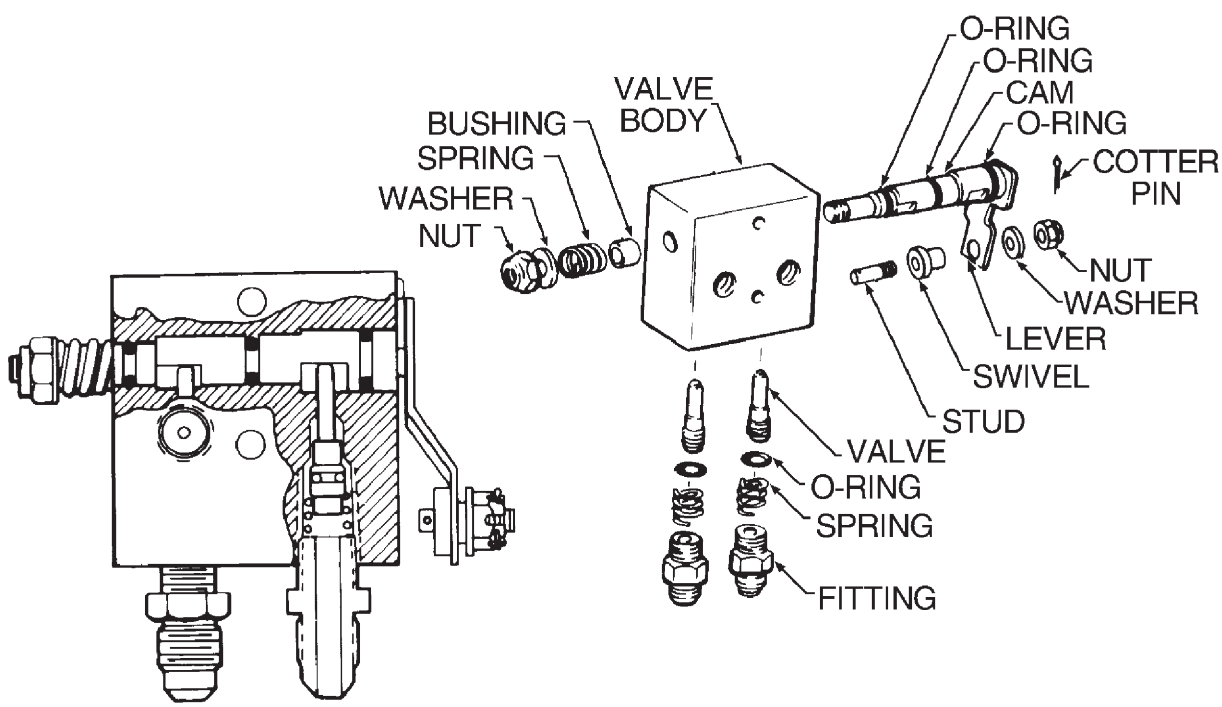
(1) Clean the valve parts with a suitable solvent and dry thoroughly.

(2) Inspect valve and seat surfaces of valve body for excessive wear and corrosion.

(3) Inspect the cam assembly for burrs, scratches, excess wear, loose operating lever, etc.

(4) Check general condition of valves and springs.

(5) Repair to the valve is largely limited to smoothing burred or scratched surfaces and replacing O-rings.



Parking Brake Valve Assembly (Parker-Hannifin)
Figure 8

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STEERING

The nose gear is steered by means of the rudder pedals. Movement of the pedals is transmitted to the steering arm by two steering pushrods, a steering bellcrank and a steering bungee. Bellows are installed on the forward portion of the steering pushrods where they pass through the forward pressure bulkhead.

1. Steering Pushrod

   A. Removal

      (1) Disconnect the aft end of the pushrod from the arm on the rudder bar.

      (2) Disconnect the forward end of the pushrod from the steering bellcrank.

      (3) Loosen the forward end of the bellows and slip the pushrod through the bellows. The rod end on the forward end of the bellows will have to be removed to allow the rod to pass through the opening in the bellows.

   B. Installation

      (1) Set the length of the steering pushrods to an initial length of 11.90 inches between the centers of the bolt holes in the rod end bearings. Ensure that the rod ends are positioned 90° to each other. Loosen the forward jam nut 1/2 turn. Remove the rod end / jam nut without turning the jam nut on the rod end shaft any more.

      (2) Insert the steering rod assembly into the aft end of the bellows. Push the rod through the sleeve in the forward end of the bellows.

      (3) Reinstall the rod end on the end of the pushrod. Then tighten the rod end 1/2 turn.

      (4) Attach the forward end of the steering rod to the steering bellcrank.

      (5) Attach the aft end of the steering rod to the arm on the rudder bar.

2. Steering Bellcrank

   A. Removal

      (1) Disconnect the forward ends of both steering pushrods from the bellcrank.

      (2) Disconnect the aft end of the bungee from the left side of the bellcrank.

      (3) Cut the safety wire and remove the bolt that secures the steering bellcrank to the outer support plate.

      (4) Remove the bellcrank.

   B. Installation

      (1) Ensure that two bushings and spacer are in place in the steering bellcrank.

      (2) Place washer over bolt and slide bolt into steering bellcrank.

      (3) Place washers over bolt. Insert bolt in support plates, insuring that spacer is in place between inner and outer support plate. Torque bolt and safety bolt to clip with MS20995-C41 safety wire.

      (4) Connect the aft end of the steering bungee to the left side of the steering bellcrank.

      (5) Connect the forward ends of both steering pushrods to the bellcrank.

3. Bungee Assembly

   A. Removal

      (1) Disconnect forward end of bungee from left-hand side of steering arm.

      (2) Disconnect aft end of bungee from left-hand side of steering bellcrank.

      (3) Remove bungee.

B. Installation

(1) Place bungee in position.
(2) Connect aft end of bungee to left-hand side of steering bellcrank.
(3) Connect forward end of bungee to steering arm. The thinner washer goes under the bolt head, the thicker washer under the nut.

4. Steering Pushrod Bellows

A. Removal

(1) Disconnect steering pushrod from steering bellcrank. Remove rod end bearing and jamnut from forward end of steering pushrod.
(2) Slide pushrod aft.
(3) Remove screws from fire shield. (The fire shield is the plate on the aft side of the pressure bulkhead which helps to secure the large end of the bellows to the aft side of the pressure bulkhead.)
(4) Slide fire shield, packing ring/housing/spacer assembly and plate aft on steering rod tube.
(5) Push bellows forward out of cutout in pressure bulkhead.
(6) Remove bellows from steering rod.

B. Installation

(1) Place bellows over forward end of steering pushrod.
(2) Push large end of bellows through cutout in pressure bulkhead.
(3) Slide plate, packing ring/housing/spacer assembly and fire shield forward on steering rod tube.
(4) Line up holes in pressure bulkhead, plate and fire shield; install screws, washers and nuts.

NOTE: Install all screws, except the uppermost one, with their shanks pointing aft. The uppermost screw's shank should point forward.

(5) Install steering pushrod per instructions in this chapter.
(6) Reconnect steering rod to steering bellcrank.

NOTE: With nose landing gear pointing straight ahead, the forward end of the bellows should be approximately even with the forward end of the steering rod tube.

5. Steering Arm

A. Removal

(1) Disconnect the forward end of the bungee from the left-hand side of the steering arm.
(2) Remove the nut and washer that secures the steering arm to the mount plate.
(3) Lift the steering arm off of the bolt.

B. Installation

(1) Position the steering arm on the bolt, ensuring that a washer is installed above and below the spacer which goes through the bushing in the steering arm. Install and torque nut.
(2) Attach forward end of bungee to left side of steering arm.
(3) Adjust the clearance between rollers on the nose gear and steering arm as desribed in Rigging and Adjustment, Rudder Controls, 27-20-00.

POSITION AND WARNING

1. Troubleshooting

(See "Chart 1")

WARNING: ALWAYS PLACE THE AIRPLANE ON JACKS BEFORE ATTEMPTING ANY TROUBLESHOOTING OF THE LANDING GEAR. SEE 7-10-00.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING LANDING GEAR WARNING SYSTEMS

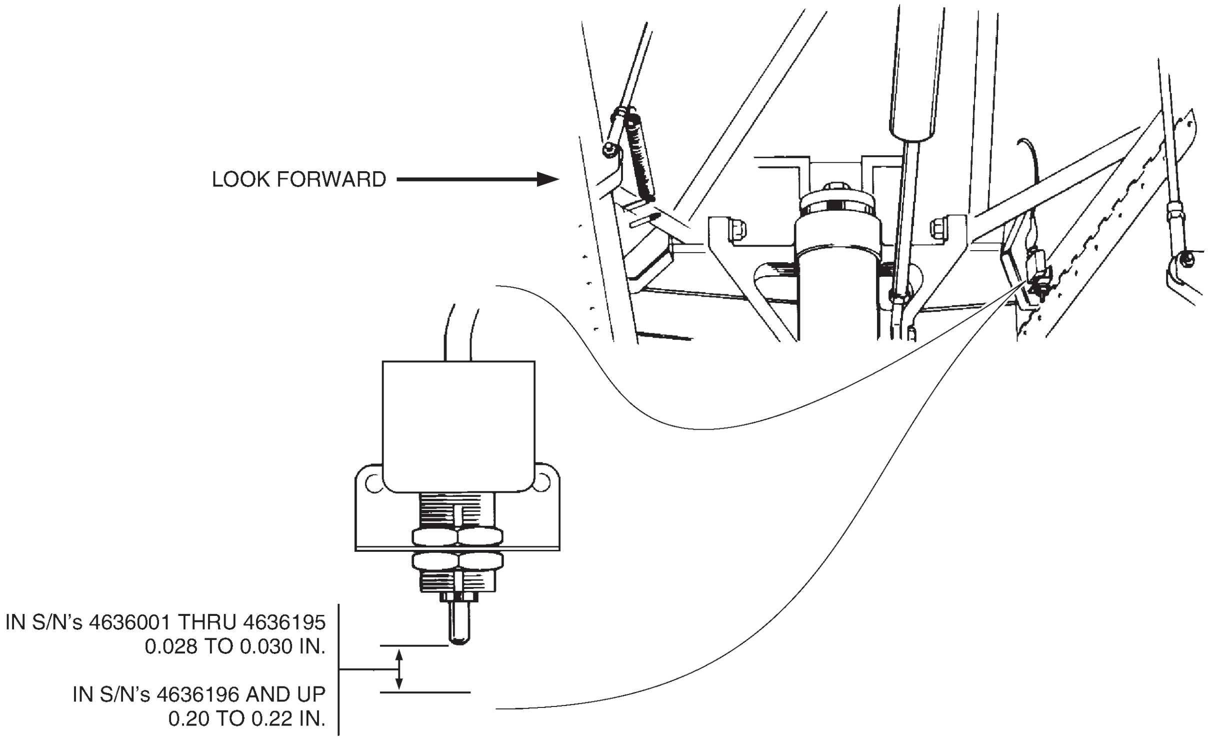
Trouble	Cause	Remedy
Red gear unsafe light out while gear is in transit.	Indicator lamp burned out.	Replace lamp.
	Indicator light ground incomplete.	Check ground circuit.
	Indicator light circuit wire broken.	Check wiring.
	Indicator light circuit breaker open.	Reset circuit breaker and determine cause for open circuit breaker.
Red gear unsafe light on though gear has retracted.	One or more up limit switches failed.	Isolate and replace switch.
	Nose gear up limit switch out of adjustment.	Check gear up adjustment and readjust up limit switch.
	Main gear not retracting far enough to actuate switch.	Check gear up adjustment.
Red gear unsafe light on though gear is down and locked.	One or more down limit switches failed.	Isolate and replace switch.
	Nose gear down limit switch out of adjustment.	Readjust down limit switch.
	Main gear down limit switch out of adjustment.	Readjust down limit switch.
NOTE: The out-of-adjustment or failed switch may be determined by noting which down light is not lit.		
Red gear unsafe light operates on and off after gear has retracted.	Light circuit wire loose.	Check wiring.
	Hydraulic system losing pressure.	Refer to Chapter 29 and Section 32-30-00.
	Gear up switch out of adjustment.	Check gear up adjustment and then switch adjustment.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING LANDING GEAR WARNING SYSTEMS

Trouble	Cause	Remedy
Red gear unsafe light out and one green gear down light out though gear is down and locked.	Lamp burned out.	Replace lamp.
	Gear down limit switch failed.	Replace switch.
NOTE: Ascertain navigation lights are off (daytime).		
	Light circuit wire broken.	Check wiring.
Red gear unsafe light and all green lights out.	Indicator lights circuit breaker open.	Reset circuit breaker and determine cause for open circuit breaker.
NOTE: Ascertain navigation lights are off (daytime).		
	Light circuit wire broken.	Check wiring.
Red gear unsafe light and horn fail to operate when power is decreased to approximately 15 IN. HG. MAP and landing gear is retracted.	Landing gear selector circuit breaker open.	Reset circuit breaker and determine cause for open circuit breaker.
	Micro switch at throttle out of adjustment.	Adjust micro switch.
	Micro switch at throttle failed .	Replace switch.
	Warning horn and light circuit wire broken.	Check wiring.
Red gear unsafe light and horn fail to stop when throttle is closed and gear has extended. (Gear extended through the use of the emergency gear down.)	Gear selector handle in up position.	Place handle in down position.
Red gear unsafe light and horn fail to operate when selector switch is moved to up position with gear extended and throttle not full forward.	Warning light and horn circuit wire broken.	Check wiring.
Above condition on ground.	Defective safety (squat) switch.	Replace switch.
Hydraulic pump shuts off, but red gear unsafe light remains on.	Gear not fully retracted.	Check gear retraction adjustments.
	Gear not contacting up micro switches.	Check gear up switches.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING LANDING GEAR WARNING SYSTEMS

Trouble	Cause	Remedy
Green gear down lights dim though position light switch is off and gear is down and locked.	Failed instrument panel light control switch. (Lights grounding through dimming resistor instead of instrument panel light control.)	Replace switch.
Green gear down light fails to go out with gear in transit or retracted.	Gear down limit switch failed.	Replace switch.
Green gear down lights will go out and not dim when position light switch is turned on though gear is down and locked.	Green light ground dimming resistor open.	Replace resistor.
Green gear down lights blink momentarily before the down lock is engaged on roller.	Micro switch out of adjustment.	Adjust micro switch.



Nose Gear Up Switch Adjustment
Figure 1

2. Nose Gear

WARNING: ADJUSTMENT OF ALL LIMIT SWITCHES SHOULD BE MADE WITH THE AIRPLANE ON JACKS.

A. Up Limit Switch Adjustment

(See "Figure 1".)

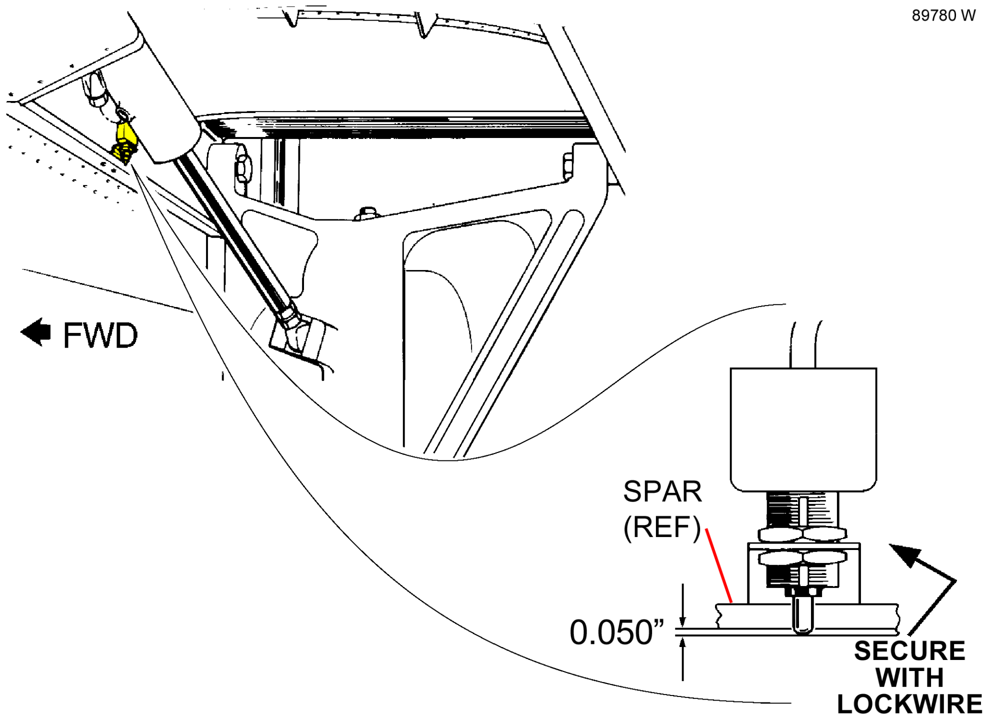
The nose gear up limit switch is located on the right-hand side of the wheel well opening. The right-hand nose wheel door actuates this switch.

This switch is adjusted by backing off the two nuts on the switch plunger bushing.

B. Down Limit Switch Adjustment

The nose gear down limit switch is mounted on the nose gear locking actuator. The switch is activated when the cylinder is extended and locked.

No adjustment of this switch is allowed.



Main Gear Up Switch Adjustment
Figure 2

3. <u>Main Gear</u>

WARNING: ADJUSTMENT OF ALL LIMIT SWITCHES SHOULD BE MADE WITH THE AIRPLANE ON JACKS.

A. Up Limit Switch Adjustment

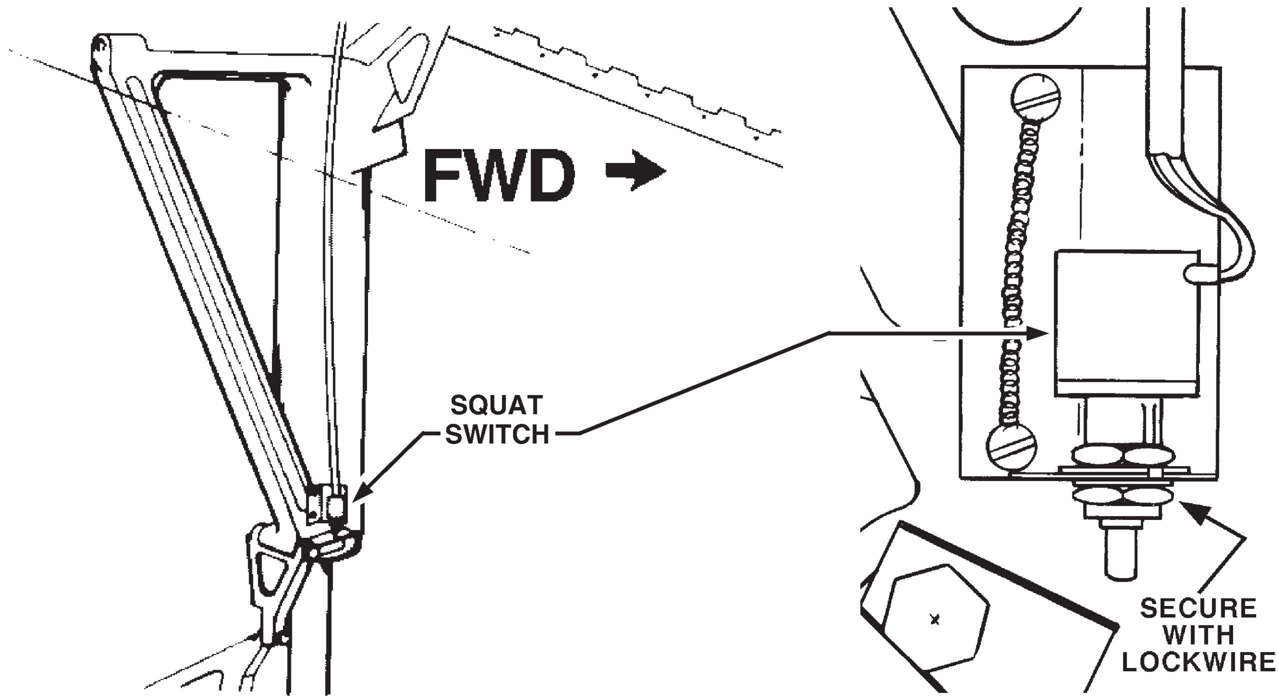
See “Figure 2”.

A gear up limit switch is located in the wing wheel well, beneath the locking actuator. There is no adjustment of the switch other than that shown in “Figure 2”; i.e. - 0.050 inch below the flush line of the spar. Secure the switch with lockwire.

B. Down Limit Switch Adjustment

The main gear down limit switches are mounted in each main gear locking actuator. The switch is activated when the cylinder is extended and locked.

No adjustment of these switches is required.



Adjustment of Squat Switch
Figure 3

4. Landing Gear Safety Switch (Squat Switch) Adjustment

(PIR-PPS60033-7, Rev. G/-9, Rev. N/-13, Rev. D.)

See “Figure 3”.

WARNING: ADJUSTMENT OF ALL LIMIT SWITCHES SHOULD BE MADE WITH THE AIRPLANE ON JACKS.

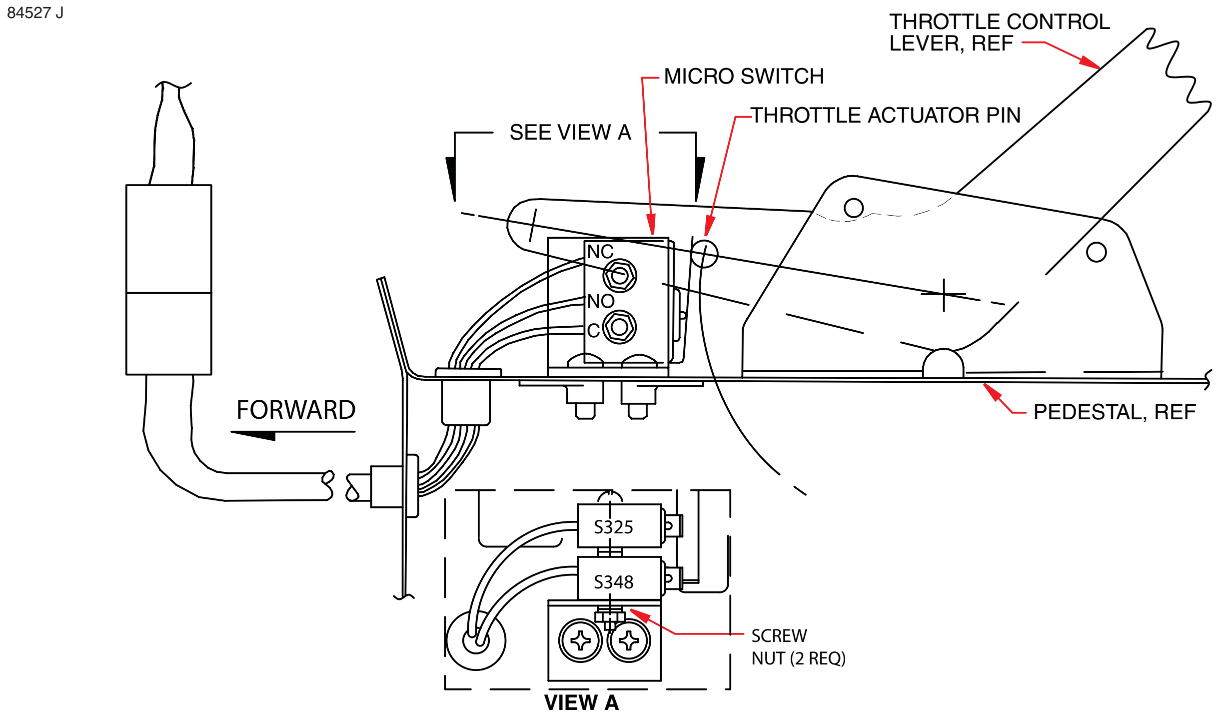
The landing gear safety switch is located on the left main gear housing. Adjust it to provide a closed gear selector “UP” control circuit at a strut extension of 8.00 inches maximum and an open gear “UP” control circuit at a strut extension of 7.70 inches minimum.

A. Jack the airplane (refer to 7-10-00).

B. Compress the strut until 7.875 inches is obtained between the top of the gear fork and the bottom of the gear housing. Hold the gear at this measurement.

C. Adjust the switch so that it actuates at this point. Secure the switch with lockwire.

D. Extend and then compress the strut to ascertain that the switch will actuate within the last quarter of an inch of oleo extension.



Landing Gear Up / Power Reduced Warning Switch
Figure 4

Effectivity
4636001 thru 4636374

5. Landing Gear Up / Power Reduced Warning Switch

See “Figure 4”.

A. S/N’s 4636001 thru 4636374

(1) Removal

The landing gear up / power reduced warning switch is located in the engine control quadrant assembly. It is on the left side of the throttle lever.

(a) Loosen the control quadrant cover and slide it up the engine control levers and out of the way.

(b) Disconnect the electrical leads to the switch at the plug.

(c) Remove screws, nuts, and washers that attach the switch to its mounting bracket.

(2) Installation

(a) Attach the switch to its mounting bracket with screws, nuts, and washers.

(b) Reconnect the electrical leads at the plug.

(c) Slide control quadrant cover down the engine control levers and secure it to the top of the console.

(d) Adjust the switch per the following instructions.

(3) Adjustment

(PIR-FTP2001-8, Rev. F., FTP2001-8-1, Rev. E., FTP2007-1, Rev. C.)

This switch is adjusted by backing off the two nuts on the switch mounting bracket to reposition the switch. Set for horn to sound at 15 ± 1 In. Hg. manifold pressure.

B. S/N’s 4636375 and up, S/N’s 4692001 and up

(PIR-FTP2001-8, Rev. F., FTP2001-8-1, Rev. E., FTP2007-1, Rev. C.)

Uses manifold pressure to operate a pressure switch to sound the gear warn horn if power is reduced to or below 15 ± 1 In. Hg with the gear up. No adjustment is required.

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CHAPTER

33

LIGHTS

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CHAPTER 33

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
33-List of Effective Pages	1	Mar 25/21			
	2	Sep 15/09			
33-Table of Contents	1	Mar 25/21			
	2	Feb 21/20			
33-10-00	1	Nov 30/17			
	2	Mar 25/21			
	3	Sep 15/09			
	4	Sep 15/09			
	5	Sep 15/09			
	6	Sep 15/09			
	7	Mar 25/21			
	8	Mar 25/21			
	9	Mar 25/21			
	10	Mar 25/21			
33-20-00	1	May 31/11			
	2	Feb 15/14			
	3	Feb 15/14			
	4	Sep 15/09			
33-30-00	1	Nov 30/17			
	2	Sep 15/09			
33-40-00	1	Nov 30/17			
	2	Nov 30/17			
	3	Nov 30/17			
	4	Nov 30/17			
	5	Nov 30/17			
	6	Nov 30/17			
	7	Nov 30/17			
	8	Nov 30/17			
	9	Nov 30/17			
	10	Nov 30/17			
	11	Nov 30/17			
	12	Nov 30/17			
	13	Nov 30/17			
	14	Nov 30/17			
	15	Nov 30/17			
	16	Feb 21/20			
	17	Feb 21/20			
	18	Feb 21/20			

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CHAPTER 33 - LIGHTS

TABLE OF CONTENTS

CHAPTER 33 - LIGHTS

TABLE OF CONTENTS (continued)

FLIGHT COMPARTMENT

All lighting is 24 Vdc except for the circuit breaker panels, which are lit by electroluminescent means and use an inverter to convert 24 Vdc input from the 5 amp INSTR PANEL circuit breaker, to 115 Vac 400 Hz.

On conventional gauge or Avidyne equipped aircraft, cockpit lighting consists of instrument panel post lights, which are controlled by a solid-state dimmer control unit. In G1000/G1000 NXi equipped aircraft, post lighting is replaced with LED lighting strips (left, center and right) in the glareshield, bezel lighting around the autopilot control panel and LED spotlights on both side panels directed toward the instrument panel. Radio lighting is controlled separately and in any case consists of integral lighting normally provided in radio equipment. Switches are back lighted for positive identification and visibility. An overhead light is located in the cockpit ceiling above each of the two front seats providing map lights and standby panel lighting for each crew member. These standby/ map lights are controlled by an ON-OFF, dimmer rheostat, located on the overhead, between the two lights. The electroluminescent lighting receives its power from the PANEL LIGHTS circuit breaker via the panel lights dimmer switch, taper block TB1 and the electroluminescent panel inverter.

The three dimmer control units are solid-state and powered through the ANNUN, AVIONICS DIMMING, and INSTR PANEL circuit breakers. The dimmer controls rheostats control the dimmer outputs for post lights or LED lighting, circuit beaker panels, switch, and avionics lighting intensity.

All panel lights, except for integral radio equipment lighting, are controlled from the dimmer control unit and dimming rheostat. Employ conventional troubleshooting methods: panel lights full bright, first replacing offending individual lamps. If not successful, measure socket voltage and repair as required. If all lamps are “OUT” check power from bus, through circuit breaker, harness, dimmer control, and dimmer output. Replace or repair as required.

NOTE: Refer to 91-33-10, 91-33-20, and 91-33-40 for schematics on the Lighting systems.

1. Dimmer Control Unit

   A. Removal

      (1) The dimmer control units are mounted on the aft right side of the forward pressure bulkhead. Access is gained from beneath the copilot’s instrument panel. The rheostat controls are mounted on the lower left side of the pilot’s instrument panel just below the main electrical switch panel.

      (2) Disconnect the electric harness connector.

      (3) Remove mounting screws and the dimmer control assembly.

   B. Installation

      (1) Position the dimmer control assembly in place and secure with the mounting screws.

      (2) Connect the electrical harness connector to dimmer assembly.

      (3) Cycle lighting to verify function.

2. Post Lights

   A. Removal

      (1) Disconnect wire from rear of socket behind panel.

      (2) Unscrew rear contact collar from rear of socket.

      (3) Remove mounting washer and nut from light assembly.

      (4) Pull assembly from panel mounting hole.

   B. Installation

      (1) Insert assembly into panel mounting hole.

      (2) Install mounting washer and nut onto light assembly.

(3) Screw rear contact collar from rear of socket.

(4) Connect wire to rear of socket behind panel.

C. Lamp Replacement

(1) Pull firmly on the post light hood and remove.

(2) The lamp will be pulled from socket and remain inside hood. Gently pry the lamp base flange from the hood to remove the lamp.

(3) Push the new lamp into the hood and push the hood back into the socket.

3. Annunciator Panel Lamp Replacement

See "Chart 1" and "Figure 1" and "Figure 2" on page 33104.

The annunciator panel, located at the top center of the instrument panel, provides systems visual warning information. A complete functional description is provided in 31-50-00. Complete electrical schematics are provided in 91-31-50.

A. In S/N's 4636001 thru 4636131, it is not necessary to remove the annunciator panel assembly to replace lamp bulbs. Simply "PUSH-IN" on the function light until it "clicks", and release pressure. The cover assembly will be partially ejected from the lamp base assembly. Pull the cover from the base and rotate to expose the lamp bulb. Replace defective bulb and reverse removal procedure. Depress "Test" function to verify lamp function.

B. In S/N's 4636132 thru 4636374, pull evenly on the annunciator panel bezel to access the lamp bulbs. An extraction tool and four spare bulbs are located behind the bezel. Use the extraction tool to replace defective bulb and reverse removal procedure. Depress "Test" function to verify lamp function.

NOTE: Test function only tests the bulbs, not the circuit.

C. In S/N's 4636375–4636459, 4636461–4636462, 4636481; and 4692001–4692133, 4692141, 4692149, 4692153, the annunciator panel is an LED unit. In the unlikely event of LED failure, the entire unit must be replaced.

D. In S/N's 4636460, 4636463–4636651 less 4636481 and 4636633; and 4692134 and up less 4692141, 4692149, 4692153; annunciation is integrated into the Garmin G1000 IAS and displayed in the PFD and MFD. There are no lamps.

4. Standby / Map Light

Standby / map lights are located in the cockpit headliner, one above each cockpit seat.

A. S/N's 4636001–4636374.

(1) Removal

(a) Remove three screws retaining map light lens.

(b) Remove lens and gasket.

(c) Twist lamp in socket and remove.

(2) Installation

(a) Insert lamp (GE 1495) in socket and twist to secure.

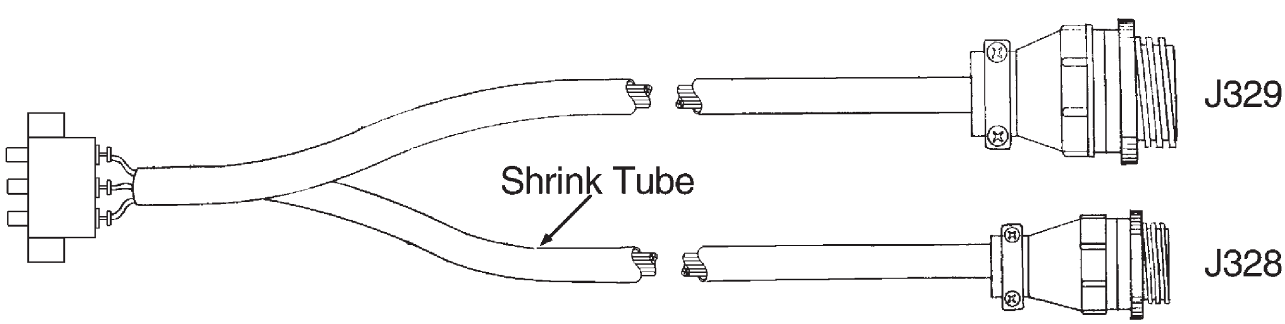
(b) Position gasket and lens and secure with three screws.

(c) Remove three screws.

CHART 1
ANNUNCIATOR PANEL LOCATOR / COLOR

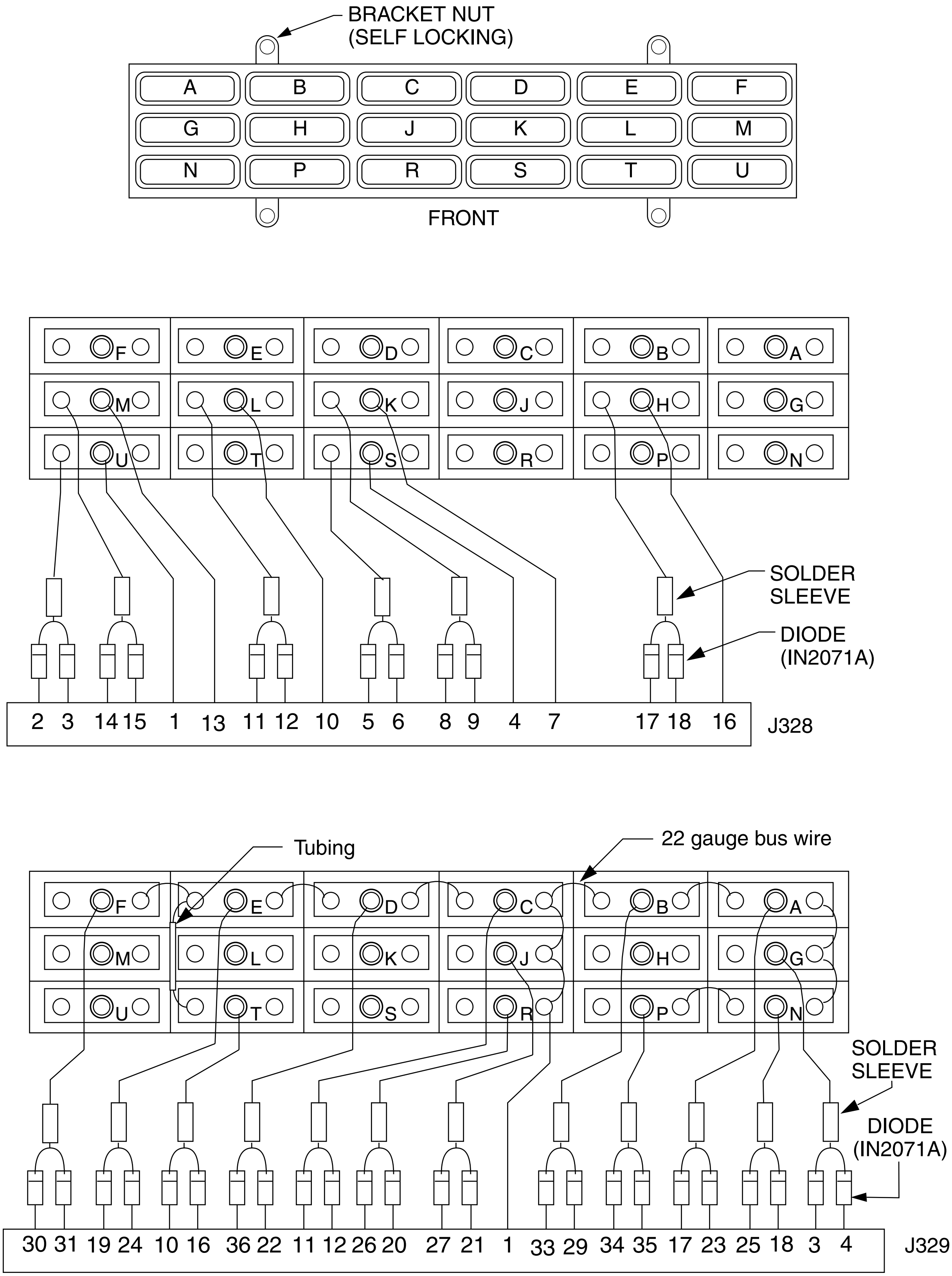
Station Fig. 2	Station Fig. 3	Annunciator	Color
A	G	ALTERNATOR NO. 1 INOP	RED
B	H	BOOST PUMP	RED
C	J	FUEL PRESS	RED
D	K	LOW BUS VOLTAGE	RED
E	L	CABIN ALTITUDE	RED
F	M	STALL WARN FAIL	RED
G	N	ALTERNATOR NO. 2 INOP	RED
H	P	GEAR WARN	RED
J	R	DOOR AJAR	RED
K	S	FLAPS	RED
L	T	STARTER ENGAGE	RED
M	U	WINDSHIELD HEAT FAIL	RED
N	V	OXYGEN	AMBER
P	--	--------------	GREEN
R	X	OIL PRESS	AMBER
S	Y	HYDRAULIC PUMP	AMBER
T	Z	SURFACE DE-ICE	GREEN
U	AA	ANNUNCIATOR INOP	AMBER
--	W	PITOT HEAT OFF / INOP	AMBER
--	A	VACUUM NO. 1 INOP	AMBER
--	B	VACUUM NO. 2 INOP	AMBER
--	C	--------------	GREEN
--	D	--------------	GREEN
--	E	--------------	GREEN
--	F	FUEL IMBALANCE	AMBER
--	--	SELECT * DE-ICE	AMBER
--	--	ICE DETECT * FAIL	AMBER

Fig. 2 = S/N's 4636001 thru 4636020 only.
Fig. 3 = S/N's 4636021 thru 4636131 only.
* = S/N's 4636132 thru 4636313, less 4636299.



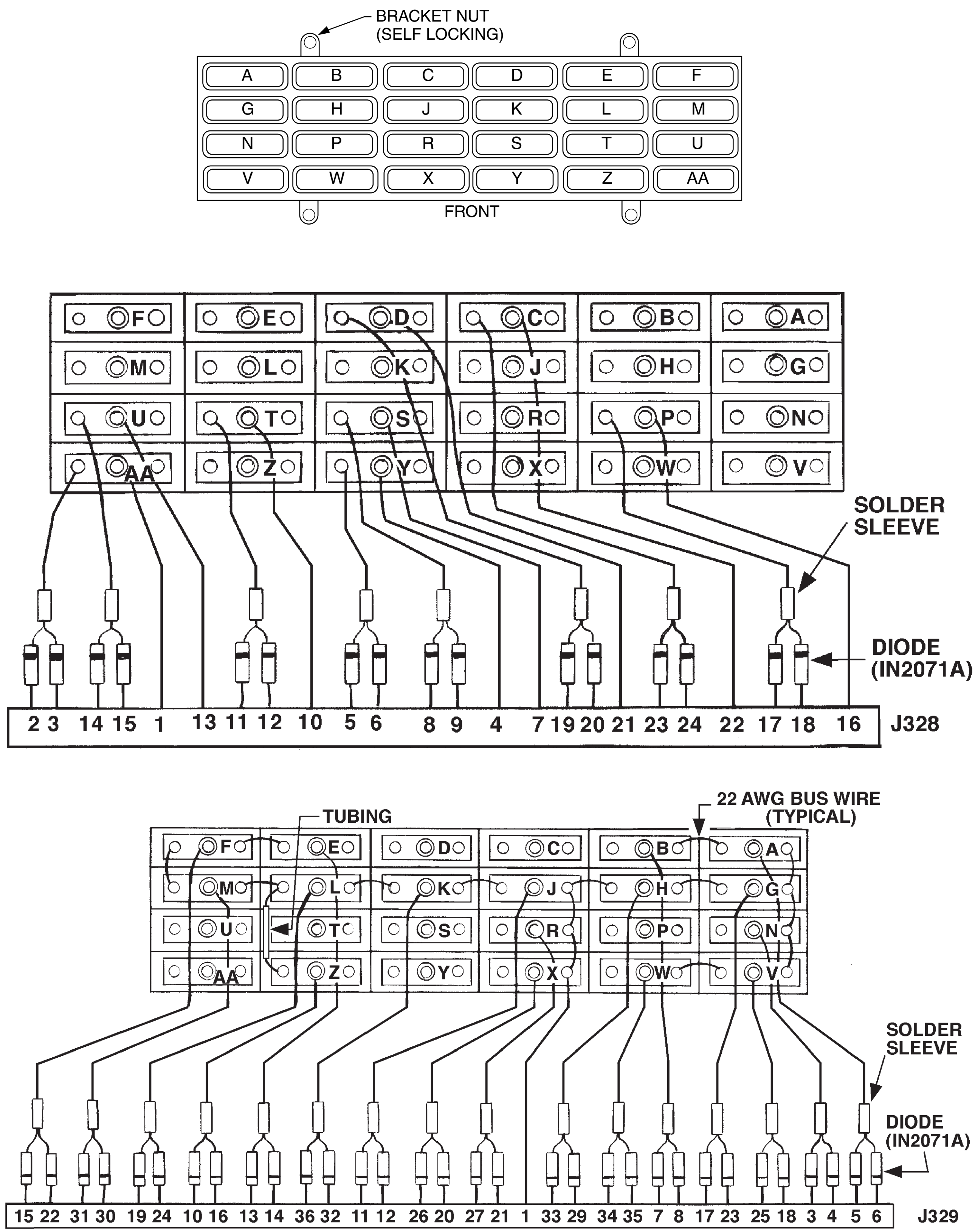
Annunciator Wiring Harness
Figure 1

Effectivity
4636001 thru 4636131



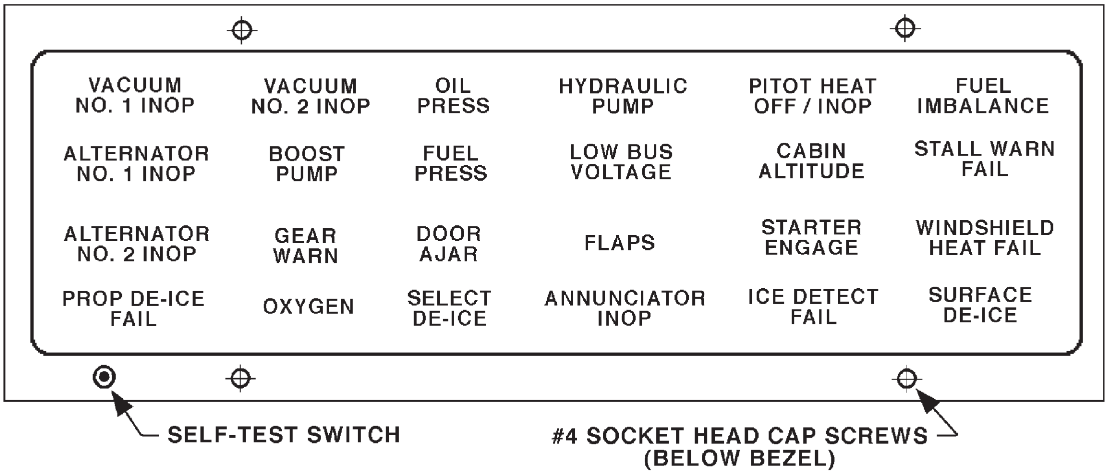
Annunciator Panel Installation
Figure 2 (Sheet 1 of 3)

33104



Annunciator Panel Installation
Figure 2 (Sheet 2 of 3)

Effectivity
4636021 thru 4636131



PIN CONNECTIONS
(37 Pin “D” Connector on Annunciator to J329 Connector)

Pin	Connection	Remarks
1	Brightness Control	From aircraft panel lights dimmer
2	Fuel Imbalance Input	Ground
3	Stall Warn Fail Input	Ground
4	Cabin Altitude Input	Ground
5	Pitot Heat OFF / Inop Input	Ground
6	Low Bus Voltage Input	Ground
7	Hydraulic Pump Input	+28VDC
13	Fuel Press Input	Ground
14	Oil Press Input	Ground
15	Vac No. 2 Input	Ground
16	Vac No. 1 Input	Ground
17	Boost Pump Input	Ground
18	Press to Test Output	+28VDC
19	Power Input	+28VDC
20	Starter Engage Input	+28VDC
21	Windshield Heat Fail Input	+28VDC
22	Surface De-ice Input	Ground
23	Ice Detect Fail Input	Ground
24	Flaps Input	+28VDC
25	Annunciator Inop Input	+28VDC
26	Case Ground	
31	Select De-ice Input	Ground
32	Door Ajar Input	Ground
33	Gear Warn Input	+28VDC
34	Oxygen Input	Ground
35	Prop De-ice Fail	Ground
36	Alt No. 2 Inop Input	Ground
37	Alt No. 1 Inop Input	Ground

Pins 8-12 and 27-30 are connected to Case Ground.

Effectivity
4636132 thru 4636313,
less 4636299 shown

Annunciator Panel Installation
Figure 2 (Sheet 3 of 3)

B. S/N's 4636375–4636775; and 4692001–4692214.

(1) Removal

(a) Remove cockpit headliner (see 25-10-00).

(b) Disconnect map light wiring harness.

(c) Remove three nuts and washers securing map light assembly to headliner.

(d) Remove map light assembly.

(2) Installation

(a) Position map light assembly in cockpit headliner.

(b) Secure map light assembly to headliner with three washers and nuts. Torque nuts not more than 12 in·lbs.

(c) Connect map light wiring harness.

(d) Reinstall cockpit headliner (see 25-10-00).

C. S/N's 4636776 and up

(1) Removal

(a) Gain access to the LED lamp module by pulling on lens holder/bezel to remove lens holder/bezel from ball housing.

(b) Unscrew LED lamp module from ball housing.

(c) Withdraw LED lamp module and harness to expose harness connector.

(d) Capture airplane side of wiring harness to ensure it does not retract above headliner.

(e) Disconnect wiring harness and remove LED lamp module.

(2) Installation

(a) Connect wiring harness to LED lamp module.

(b) Release airplane side of wiring harness and feed wiring harness through ball housing until LED lamp module engages threads in ball housing.

(c) Screw LED lamp module from ball housing.

(d) Reinstall lens holder/bezel on ball housing.

5. Dimming Rheostat

A. Removal

(1) Remove the overhead access panel between pilot's standby map light and radio speaker.

(2) Remove rheostat mounting screws.

(3) Disconnect and label wiring.

(4) Remove rheostat.

B. Installation

(1) Connect wiring to rheostat.

(2) Position rheostat in proper location and install mounting screws.

(3) Install the overhead access panel.

6. Switch Lamps Removal and Installation

   A. On the exposed arc of the rocker switch find an indentation and pry the rocker portion of switch from the switch assembly, using the indentation for leverage, note rocker position.

   B. Locate and pull the metal tab. Remove lamp from its socket and replace. It is not necessary to rotate lamp when removing or installing it.

   C. After replacing the lamp, place the rocker portion of the switch in the same position as when removed, and press it firmly into place noting an audible “snap” as the plastic locks mate into place. Cycle the switch to confirm rocker is seated.

7. Illuminated Switch Assembly Removal and Installation

   CAUTION: ALTHOUGH SMALL SWITCH ASSEMBLIES ARE FAR EASIER TO REMOVE IF WIRING IS FIRST DISCONNECTED, THE LIMITED WORK SPACE BEHIND THE PANEL CAN RESULT IN BURNED WIRE INSULATION. DO NOT ATTEMPT TO UNSOLDER THESE SMALL ELECTRICAL CONNECTIONS BEHIND THE INSTRUMENT PANEL UNDER ANY CONDITIONS. IF NECESSARY CUT THE WIRES AT THE POINT OF CONNECTION. IN ANY CASE IT IS BETTER TO DAMAGE THE SWITCH AT TIME OF REMOVAL AND REPLACE IT, RATHER THAN DAMAGE THE WIRING HARNESS LEADS.

   A. To remove, squeeze metal lock tabs at rear of panel and push switch from the rear sliding assembly from the front of panel.

   B. To install, align the switch from the panel front and push firmly into the panel cutout until the metal lock tabs snap into position against the rear face of the panel. Make sure the electrical connections are completed first.

8. Environmental/De-ice Switch Panel

   S/N’s 4636132 and up and S/N’s 4692001 and up.

   The Environmental/De-ice switch panel adjacent to the annunciator, in S/N’s 4636132 and up and S/N’s 4692001 and up, is a sealed unit lighted with LED’s and cannot be repaired in the field. Return failed units to Piper for repair or replacement.

9. Circuit Breaker Panels Electroluminescent Lighting

   Illumination by electroluminescence is accomplished by discharging an alternating current through phosphor treated material, such as the circuit breaker decal strips. Should any one decal strip fail to illuminate, the complete strip must be replaced.

   Should all decal strips fail to illuminate, the probable fault is with the inverter used to convert 24 Vdc to 115 Vac. The inverter is located on the rib between the pilot’s circuit breaker panels, below the panels. Replace the inverter.

   The 24 Vdc power to the inverter comes through the INSTR PANEL 5 amp circuit breaker. If the electroluminescence placards and post lights fail to illuminate, check that the INSTR PANEL circuit breaker has not tripped; reset, if necessary. If the INSTR PANEL circuit breaker has not tripped, check the PANEL dimmer control unit.

   If the PANEL dimmer control unit is good, check that neither MAIN 60 amp Tie Bus circuit breaker has tripped; reset as necessary.

   If none of the above correct the problem, check all related wiring. Refer to Chapter 91 for appropriate electrical schematics.

10. Post-Installation Set-up

The GNS-430/-530 NAV/COM/GPS and the GTX-327/-330 Transponder each have lighting configuration settings which must be adjusted after installation. Whenever a unit is replaced, use the procedures found in 34-50-00, Post-Installation Set-up Procedure, to configure the lighting in the new unit.

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PASSENGER COMPARTMENT

1. Courtesy / Rear Baggage Area Lighting

A. Description

Cabin courtesy, rear baggage area and forward baggage compartment lighting is powered through an in-line 5 amp fuse from the battery bus. Two switches, an upper and lower, are actuated by the cabin entrance door. One half of the lower switch controls the cabin courtesy lighting, while the other half and the upper portion of the switch provides door ajar warning power to the annunciator panel. The cabin entrance lighting circuit is controlled by a five minute “one-shot” timer to prevent unnecessary battery discharge if the door is left open unattended. The lights are located to ensure the entrance steps are illuminated. The forward baggage compartment light is controlled by a switch in the forward baggage compartment that is activated when the baggage compartment door is opened.

NOTE: All lighting is 24 Vdc.

B. Lamp Replacement

(1) Remove lens mounting screws.

(2) Pull out bayonet base lamp.

(3) Install replacement, check, and reinstall lens.

2. Courtesy Lights Timer

A. Removal

(1) The courtesy lights timer is located on a frame assembly aft of the lower cabin door.

(2) Disconnect electrical wires.

(3) Remove mounting screws and timer assembly.

B. Installation

(1) Position timer assembly in place and secure with mounting screws.

(2) Connect electrical wires.

(3) Test for proper operation of timer.

3. Reading Lights and Lamp

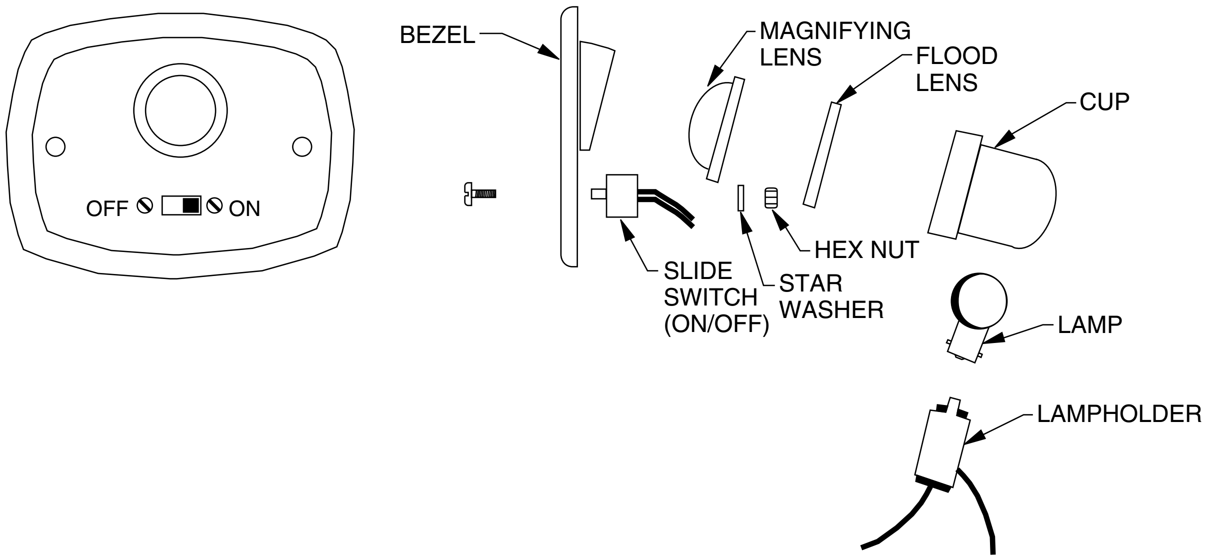
A. Removal

(1) Remove two faceplate mounting screws and gently pull bezel from ceiling cavity, pulling switch and lamp wires into cabin area until lamp holder is exposed. Be careful magnifying and flood lenses do not drop to floor (refer to “Figure 1”).

(2) Squeeze lamp holder and withdraw it from cup. Replace bayonet lamp.

NOTE: If equipped with LED light, the entire lamp assembly must be replaced in the event of a failure.

(3) Slide switch can also be replaced in bezel after removal.



Reading and Courtesy Light
Figure 1

B. Installation

(1) With new lamp installed in lamp holder, squeeze lamp holder mount tabs and insert holder into cup. Make sure lamp holder locks securely in place.

(2) Place magnifying lens and flood lens in position on bezel, holding it horizontal, and place cup in position on bezel.

(3) Holding assembly together, insert entire assembly into ceiling cavity.

(4) Align faceplate mounting holes and reinstall faceplate mounting screws.

4. <u>Side Light Strips</u>

The side light strips are illuminated by hundreds of miniature light bulbs within each strip. Bulbs cannot be replaced individually. The side light strips are controlled by a switch adjacent to the cabin door.

A. Removal

(1) The light strip assembly is located on side panel installation (refer to “Figure 2”). If required, remove the Aft Side and Cabin Side Window Panels as described in 25-20-00.

(2) Disconnect electrical wires.

(3) Remove light strip assembly.

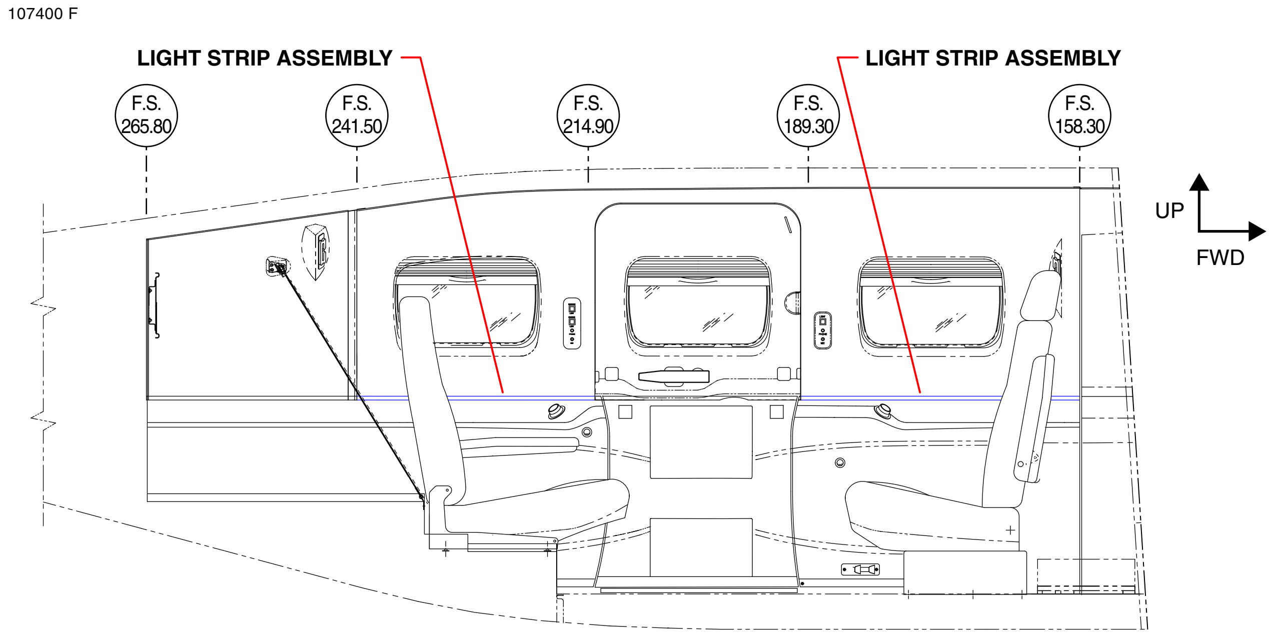
B. Installation

(1) On the back side of light assembly apply a minimum amount of RTV 732 (279-194) to hold assembly in place.

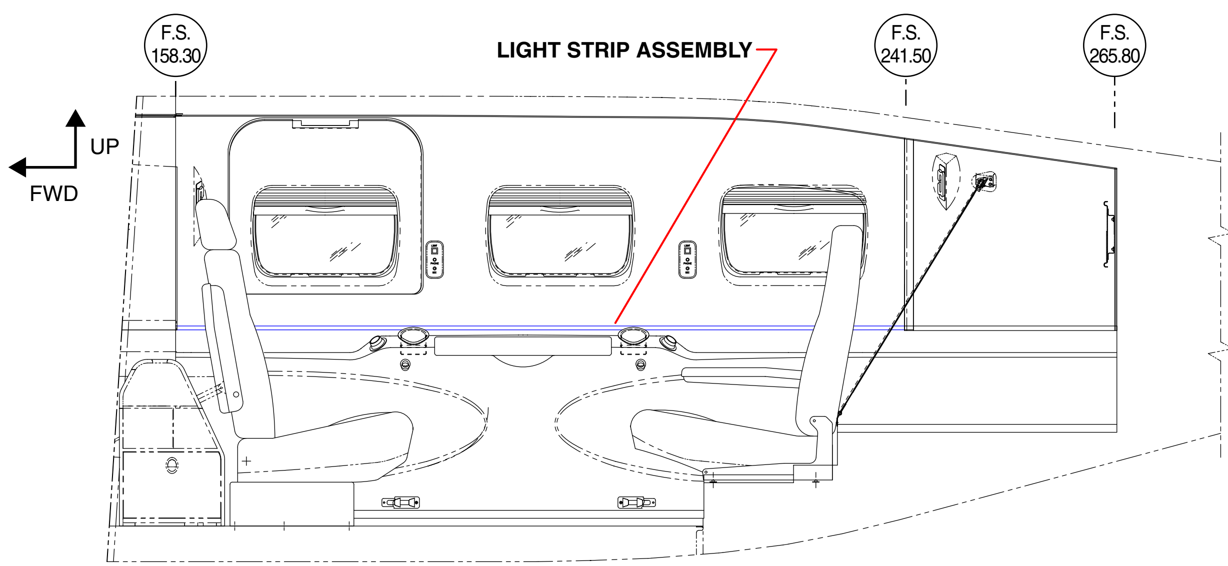
(2) Position light assembly in track and secure with masking tape to retain assembly until RTV is cured then remove tape and clean as required.

(3) Connect electrical wires.

(4) If required, install the Cabin Side Window and Aft Side Panels as described in 25-20-00.



LEFT SIDE PANEL
LH SIDE - LOOKING OUTBOARD



RIGHT SIDE PANEL
RH SIDE - LOOKING OUTBOARD

Side Light Strips
Figure 2

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CARGO AND SERVICE COMPARTMENTS

The forward baggage compartment door actuates the door ajar warning / baggage compartment light switch. The baggage compartment light may contain its own integral switch.

NOTE: All lighting is 24 Vdc.

Forward Baggage Compartment Lamp Replacement

A. Remove lens.

B. Replace bayonet lamp.

C. Replace lens and check operation.

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EXTERIOR

External lighting consists of anti-collision strobes, landing, taxi-pulse, position, and wing inspection lights. Refer to 91-33-40 for electrical schematic details of these lighting systems.

1. Landing Light

A landing light is attached to the nose gear strut by a support bracket and brace. It functions independently of the taxi lights. (See “Figure 1” on page 33402.)

A. Removal

(1) Remove the four lens attachment screws and pull the bezel from the fixture being careful not to damage the seal. Remove seal and bezel.

(2) The lamp will be free to separate from the support bracket. Pull the lamp forward and unscrew the electrical connections from the lamp.

B. Installation

(1) Replace the lamp making sure the electrical connections are secured with the electrical connection screws previously removed, and push the lamp into the lamp support bracket.

(2) Reinstall the seal and bezel with the four attachment screws.

NOTE: If equipped with an LED lamp and the new lamp does not illuminate, remove power and reverse the electrical leads to the lamp.

2. Taxi / Pulse (Recognition) Lights

A taxi / pulse light is located within the leading edge of each wing inboard of the tip. See “Figure 2” on page 33403. In S/N’s 4636132 and up, the two-position “OFF-Taxi” switch is replaced with a three-position “Taxi-OFF-Pulse” rocker switch. When switched to “Pulse”, a flasher unit (mounted under the pilot’s feet at F.S. 108.00) is engaged and the recognition/taxi lights illuminate alternately at about 55 pulses per minute.

Replacement

(1) If the strobes have just been in operation, wait five minutes for high voltage bleed down to occur before removing wing access door located on the lower surface of the wing just inboard of the wing tip by removing the 12 screws securing to the wing. The strobe power supply is attached to this wing access door with 4 screws an eight washers.

(2) Remove wing access door. If equipped with a strobe power supply, unplug power supply connectors, so access door with power supply still attached can be set aside. This will prevent weight of power supply from pulling on electrical harness. (See “Figure 3” on page 33405.)

(3) Disconnect electrical connectors from rear of light.

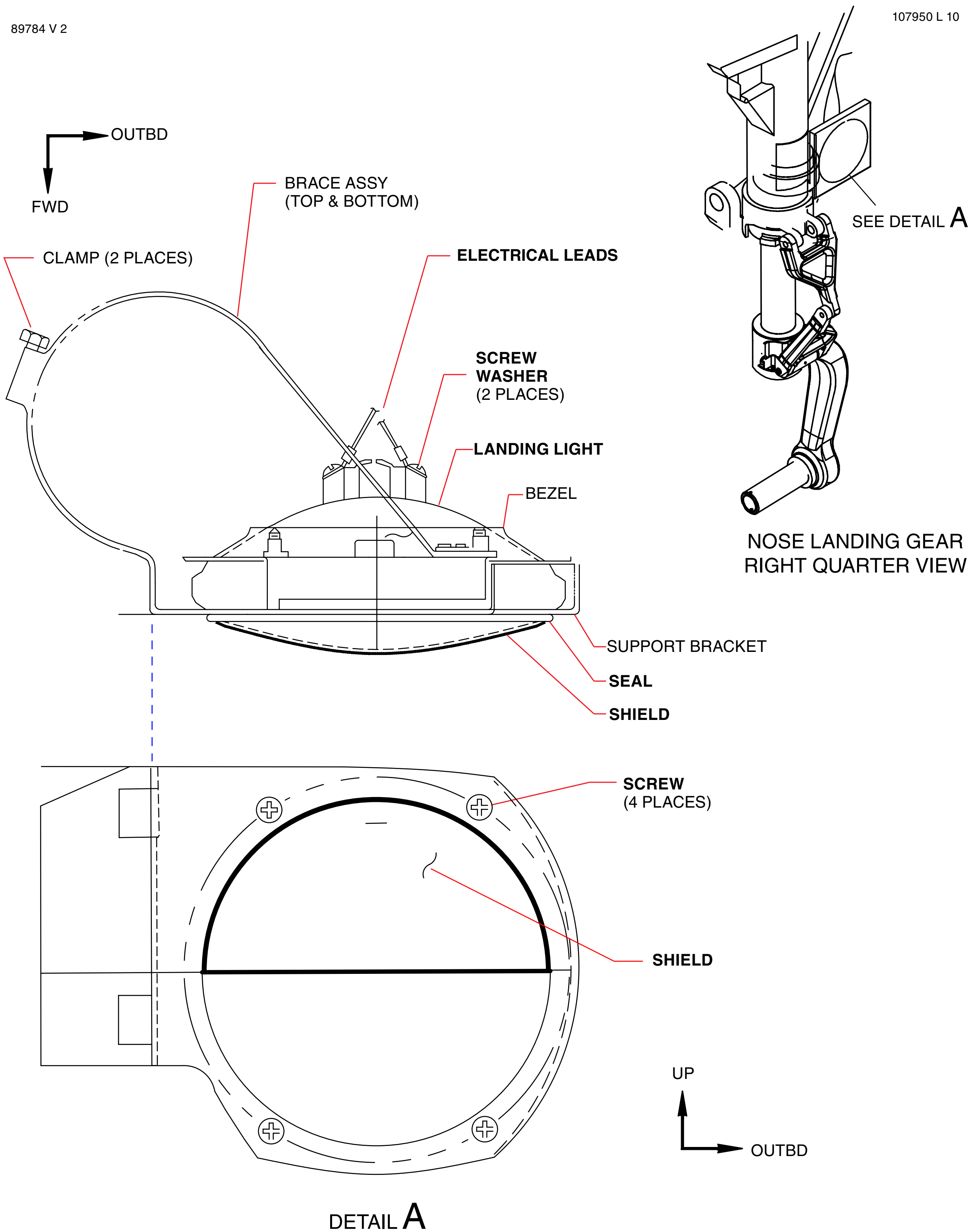
(4) Remove four bracket retaining screws from the assembly. These screws are removed from the external surface of the wing; two on the bottom and two on top.

(5) Slide the light bracket assembly aft and down and out through access opening.

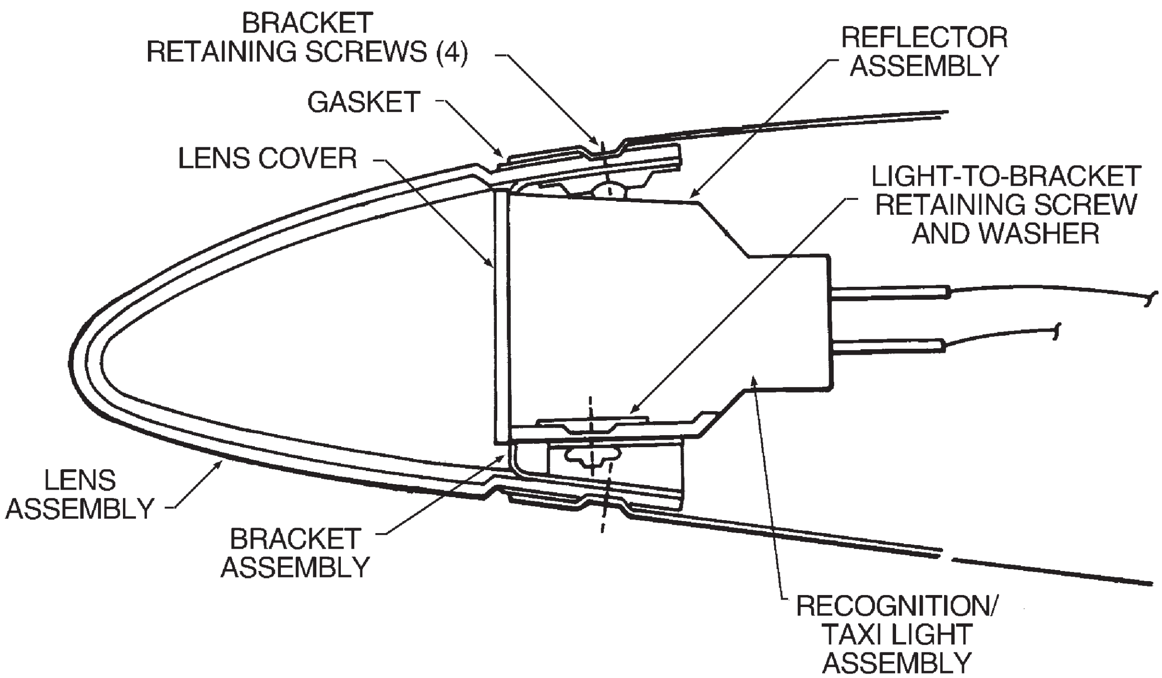
(6) To remove lens, squeeze lens gently and slide forward, out of edge of wing structure, ensuring the gasket does not slide off and into the wing.

(7) To remove lamp from conventional light assembly, snap lens cover off of the reflector and replace the lamp. If equipped with LED light, the entire lamp assembly must be replaced in the event of a failure.

(8) Installation is reverse of preceding steps.



Landing Light
Figure 1



Recognition / Taxi Light
Figure 2

3. Strobe Lights

A. PA-46-350P S/N's 4636001–4636626 and PA-46R-350T S/N's 4692001–4692208.

See "Figure 3" and 91-33-40.

(1) Description

Each wing tip strobe tube is powered by its individual power supply. Both power supplies receive power from the main bus through the 5 amp STROBE LIGHTS circuit breaker, and are controlled by the strobe light switch. If both strobe lights exhibit a fault, first verify that the STROBE LIGHTS circuit breaker has not opened (reset, if necessary), and then check integrity of the strobe light switch (replace if required). If either strobe is operating correctly, the strobe switch is considered good. Check wiring from the strobe light switch to each strobe power supply to verify system wiring integrity.

The strobe light assembly functions as a capacitor high-voltage discharge system. An audible tone of 1 to 1.5 K Hertz is emitted during the charge cycle of the strobe power supply in a normally functioning unit. However, the audible tone does not necessarily mean the power supply firing circuit is good. A voltage of approximately 450 volts is developed on the capacitor, after which the audible tone ceases. The capacitor is paralleled across the Xenon flash tube, which holds off the voltage until the tube is triggered by a separate pulse generated by a solid-state timing circuit.

After the trigger pulse causes ionization of the Xenon gas in the envelope, an intense light is given off for a brief moment as the capacitor discharges. Following discharge across the tube, the oscillator again chops up the dc aircraft current into alternating current, and again develops a higher voltage through transformer action. After full voltage is again impressed on the capacitor, the cycle repeats itself, occurring at about 50 flashes per minute.

Since each wing tip strobe is powered by its own power supply, and if both power supplies are emitting audible tones, simply switch power supplies from each wing tip. If the trouble is one of the power supplies, replace it. If the faulty strobe still malfunctions, replace that strobe tube. Refer to Replacement of Position / Strobe Lamps and to Replacement of Strobe Power Supplies.

(2) Strobe Xenon Tubes and Power Supply

General Service Information

(e) Never reverse power supply input connections for even an instant. Reverse polarity will permanently damage the circuitry. Even though damage may not be immediately apparent, the power supply will fail in use.

(f) Wait five minutes for high-voltage bleed-down to occur, before handling a strobe power supply to avoid electrical shock.

(g) Never allow connection between pins 1 and 2 of the flash tube connector, since this will discharge the capacitor and destroy the trigger circuit in the power supply.

NOTE: All of the following conditions, except item (f), are considered to require replacement.

(h) A Xenon flash tube can be photo-sensitive. Some may flash normally when exposed to light, but may become difficult to fire in darkness.

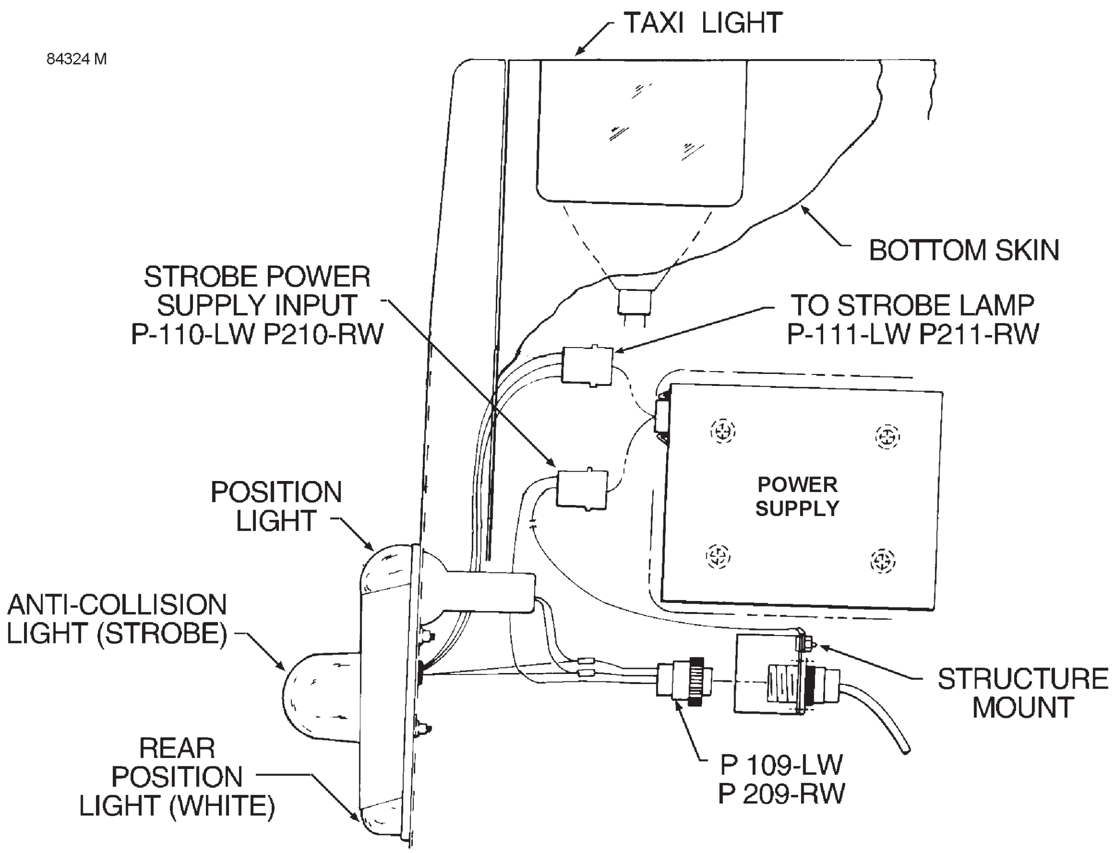
(i) Xenon flash tubes become more difficult to fire with age, or when exposed to very high temperatures.

(j) A tube with most of the service life left in it, should fire on a reduced voltage of 22 volts.

(k) A tube with little service life left in it, will fire with an engine running, but refuse to fire on battery voltage.

(l) Under hot and cold cycling during normal operation, "egg-shelling" of the glass envelope can occur, or leaks may develop in the glass seal.

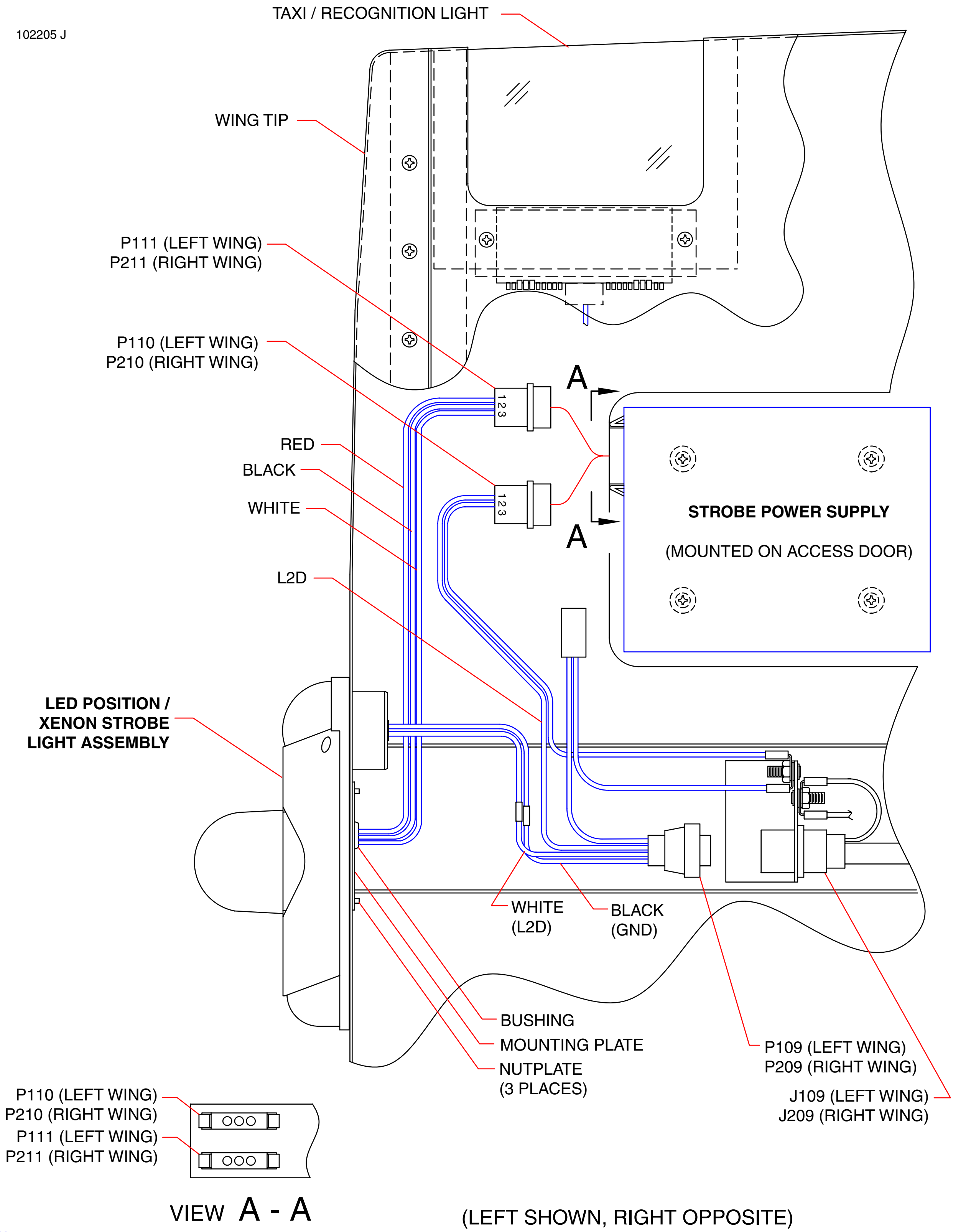
(m) Xenon strobe tubes can sometimes go into "self-ionization" continuously glowing a light blue, thus rendering the system inoperative.



(LEFT SHOWN, RIGHT OPPOSITE)

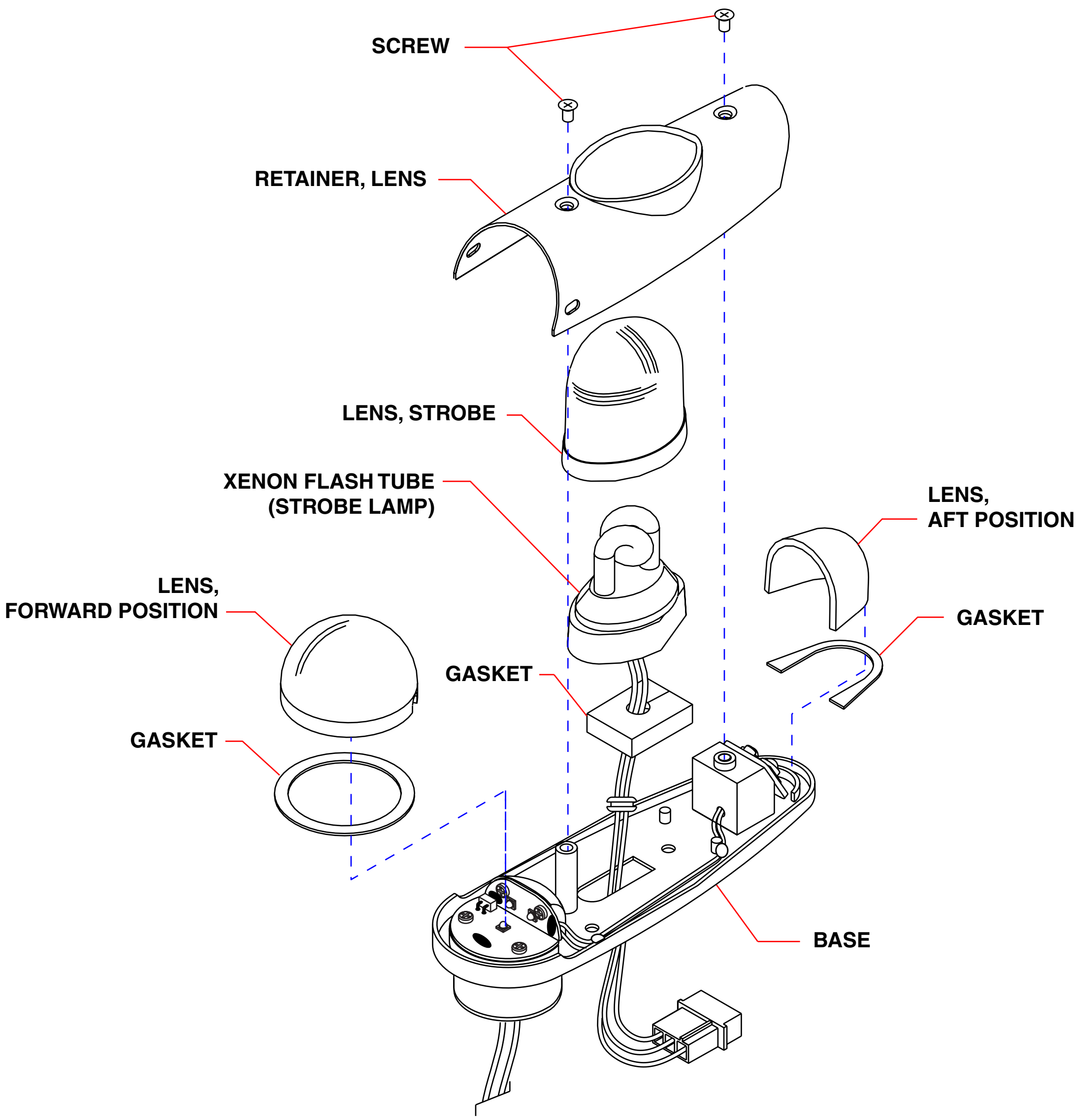
Position, Strobe, and Taxi Light Installation
Figure 3 (Sheet 1 of 3)

Effectivity
4636001–4636502
4692001–4691165



Effectivity
4636503–4636626
4692166–4691208
or with Kit 88508-002

Position, Strobe, and Taxi Light Installation
Figure 3 (Sheet 2 of 3)



Position, Strobe, and Taxi Light Installation
Figure 3 (Sheet 3 of 3)

Effectivity
4636503–4636626
4692166–4691208
or with Kit 88508-002

(3) Position / Strobe (Xenon) Lamp Replacement

See "Figure 3" on page 33405.

NOTE: In PA-46-350P S/N's 4636503–4636626 and PA-46R-350T S/N's 4692166–4692208 as delivered from the factory; and in S/N's 4636001-4636502 and S/N's 4692001-4692165 with Piper Kit No. 88508-002 installed per Piper Service Letter No. 1157; the position lights are LED and their failure requires replacement of the entire Position/Strobe Light Assembly. The strobe remains a xenon lamp that can be replaced individually.

(a) Remove the screws securing the plexiglass cover to the wing tip structure, and separate the cover from the wing tip.

(b) Remove lens retainer screws, retainer, and lens, exposing the light bracket assembly.

(c) The position lamps can now be replaced. (Not applicable to LED installations.)

NOTE: If strobe (Xenon) lamp is to be replaced, proceed to step (e). If not complete step (d).

(d) Install lens, retainer, and cover in reverse of the preceding steps.

(e) Each strobe lamp is powered by its own power supply which is attached to a wing access panel located on the lower wing surface just inboard of each wing tip.

CAUTION: WHEN DISCONNECTING STROBE POWER SUPPLY, ALLOW FIVE MINUTES OF BLEED-DOWN TIME BEFORE HANDLING A CHARGED UNIT.

(f) Remove the 12 screws securing wing access panel to the underside of the wing. Drop access panel from the wing exposing the attached strobe power supply.

(g) Remove four attachment screws and eight AN960-10 washers; separate power supply from access panel. Set wing access panel aside to protect paint etc.

(h) Unplug both power supply connectors, and remove power supply from wing.

(i) Reach into wing and unscrew (P109 left wing / P209 right wing) releasing this connector.

(j) Remove light bracket assembly mounting screws from wing tip and gently pull entire light assembly free of wing tip, feeding wiring through wing tip opening until plug (P101 / P201) three wire connector is pulled outside of wing.

NOTE: If sufficient wire length is not available to continue with the following steps, the remaining ground wire will have to be disconnected at the plug bracket assembly ground lug.

(k) Cut the disconnected three wire connector, P111 or P211, at the base of the strobe light bracket assembly and discard cut wires and connector end.

(l) Remove and destroy Xenon strobe lamp.

(m) Route new wires attached to replacement strobe lamp through hole in bracket assembly and install new Xenon tube in bracket assembly.

(n) Referring to 91-33-40 schematic of anti-collision strobe lighting insert new wiring terminals into the replacement connector plug supplied with the new strobe lamp. Wire according to schematic.

(o) Gently feed cable assembly back into wing and align light bracket assembly in place. Do not mount assembly.

(p) If ground wire was disconnected from plug bracket ground lug, connect this wire.

(q) Reconnect (P109 or P209) to plug bracket mate.

(r) Connect new three wire connector P111 or P211 and the original P110 or P210 connectors to power supply.

(s) Install power supply to access panel with the eight washers and four attachment screws.

(t) Install wing access panel with power supply attached and wires facing outboard, to the wing surface with appropriate mounting screws previously removed.

(u) Install the light bracket assembly, lens, lens retainer and plexiglass cover.

(4) Strobe Power Supply

See “Figure 3” on page 33405.

Each wing tip strobe light is powered by its individual power supply which is attached to a wing access door located on the lower wing surface just inboard of the wing tip.

(a) Removal

1) Remove wing access door by removing its twelve (12) attaching screws.

2) Unplug power supply connectors P111 and P110 on left wing and P211 and P210 on right wing.

3) Remove four screws and eight washers attaching power supply to wing access door and separate access door from power supply.

(b) Installation

1) Place power supply in position on the wing access door. Secure with the eight washers and four screws.

2) Plugin power supply connectors P111 and P110 on left wing and P211 and P210 on right wing as shown in “Figure 3” on page 33405.

3) Place wing access door in position on wing and secure with twelve (12) screws.

B. PA-46-350P S/N’s 4636627 and up and PA-46R-350T S/N’s 4692209 and up.

See “Figure 4” on page 334010 and 91-33-40.

(1) Description

These airplanes are equipped with an all LED position / strobe light assembly. If one LED fails the entire light assembly must be replaced. Access to the position / strobe light electrical harness and connectors is through the wing access door in the lower wing skin.

(2) Removal

(a) Lens

1) Remove two (2) socket head screws from aft end of light assembly and remove lens retainer.

2) Slide lens aft approximately 1/2 inch and lift away from light assembly.

(b) Light Assembly

1) Remove wing access door by removing its twelve (12) attaching screws.

2) Reach inside the wing and disconnect the P109/P209 connector.

3) Remove lens as described above.

4) Remove three (3) phillips head screws and separate light assembly from baseplate.

5) Pull light assembly away from wing tip to draw most of the attached wiring harness out. Capture the exposed wiring harness at the baseplate with tape to ensure it does not retract into the wing.

6) Cut wires close to light assembly and remove light assembly.

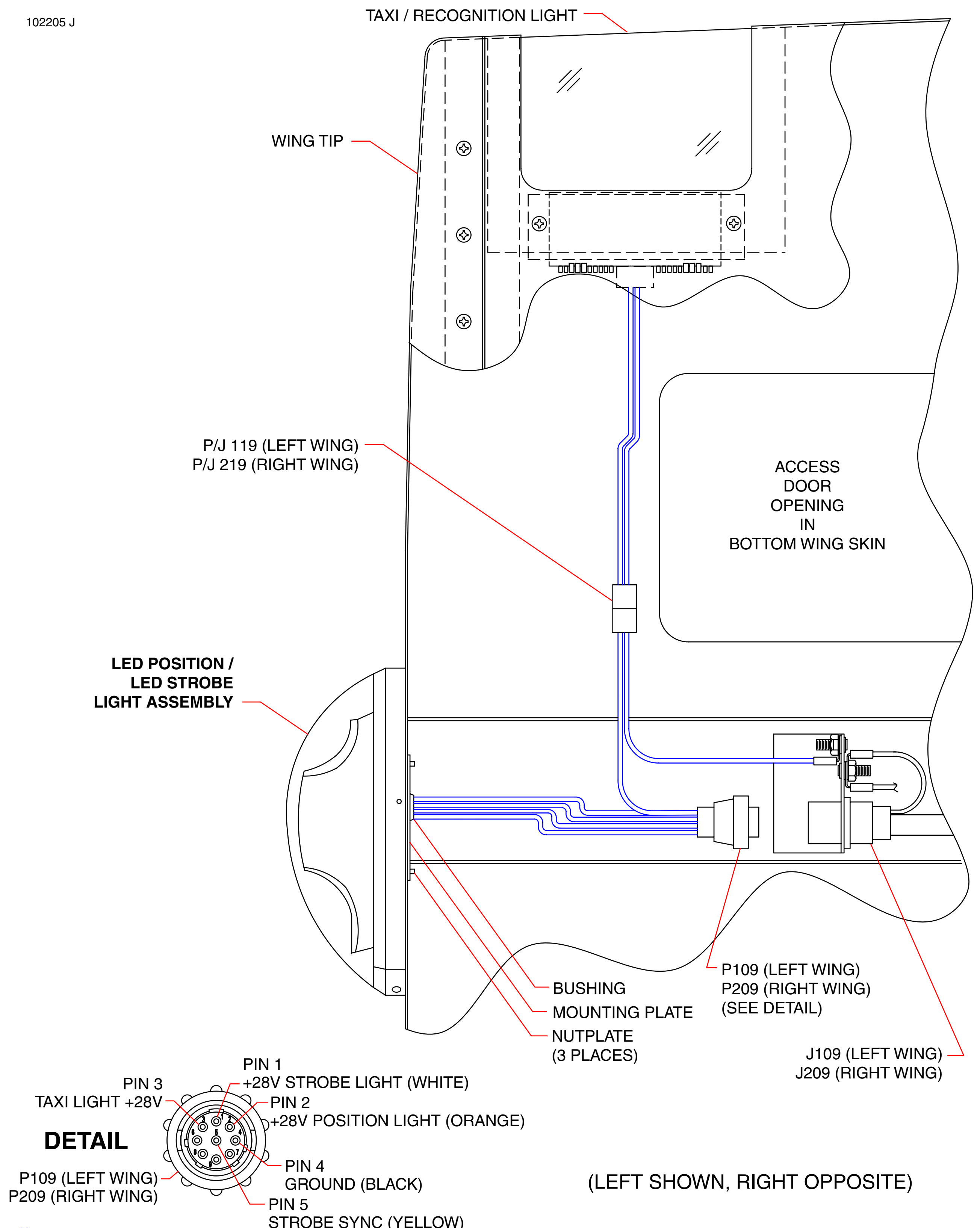
(c) Baseplate

Not required unless the existing baseplate is damaged or deteriorated.

1) Remove Lens and Light Assembly as describe above.

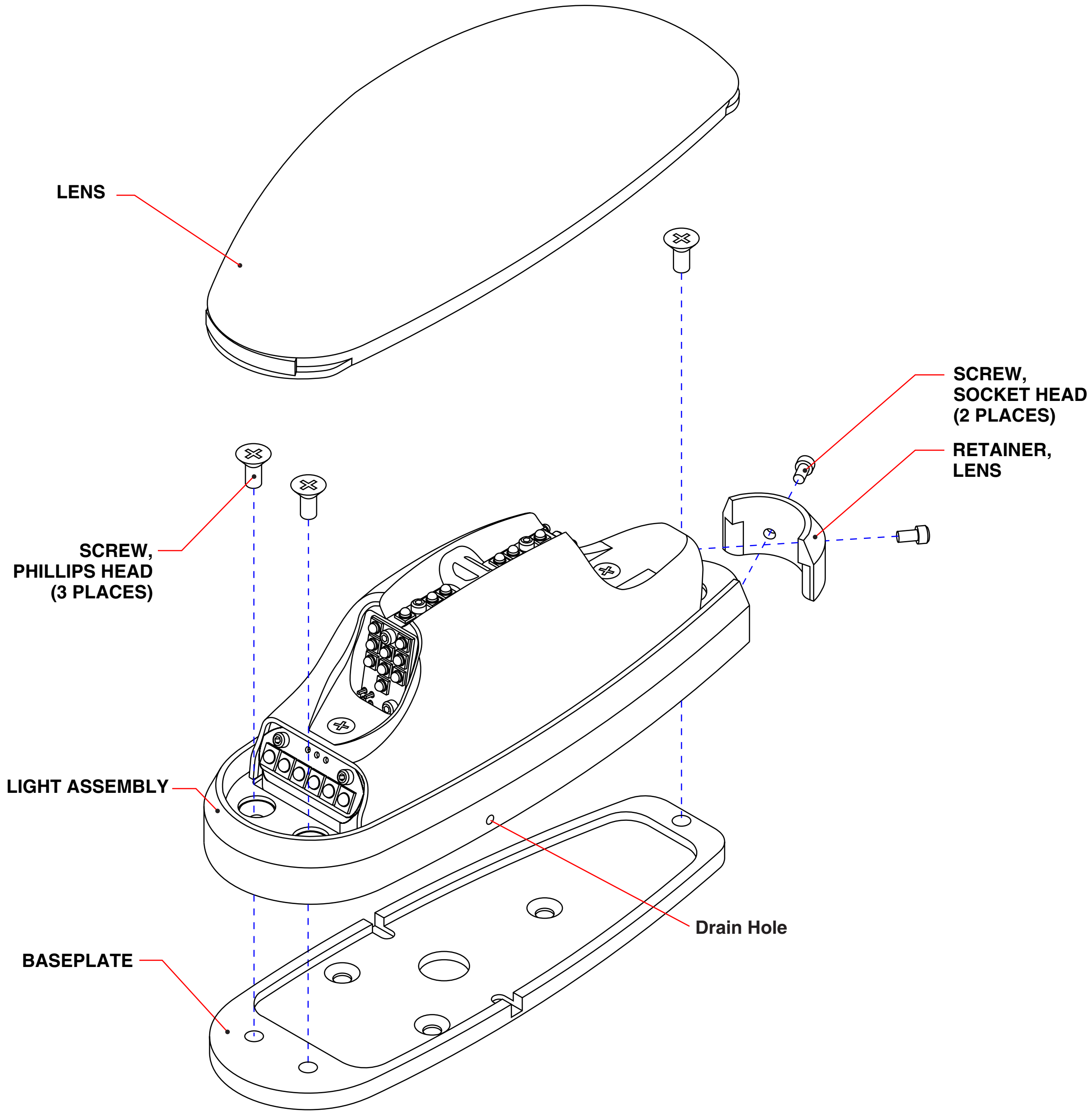
2) Remove three screws securing baseplate to wingtip and remove baseplate.

3) Remove any remaining sealant from wingtip.



Effectivity
4636627 and up
4692209 and up

LED Position and LED Strobe Light Installation
Figure 4 (Sheet 1 of 2)



LED Position and LED Strobe Light Installation
Figure 4 (Sheet 2 of 2)

Effectivity
4636627 and up
4692209 and up

(3) Installation

(a) Baseplate

1) Ensure wingtip is clean and dry.
2) Feed wiring harness through hole in baseplate and position baseplate on wingtip.
3) Secure with three screws.
4) Fillet seal around the mating surfaces of the wingtip and baseplate to prevent water intrusion. Seal with Sealant, Airframe and Component per the Consumable Materials chart in 91-10-00.

(b) Light Assembly

NOTE: If baseplate was removed, install new baseplate as described below before proceeding.

1) Splice wires from new postion/strobe light assembly to the exposed harness wires at the baseplate per the detail in “Figure 4” on page 334010.
2) Feed the wiring harness through the hole in the baseplate and place the new light assembly in position.
3) Secure with three phillips head screws.
4) Install lens as described above.
5) Reach inside wingtip and connect P109/P209 connector.
6) Place wing access door in position and secure with twelve (12) screws.

(c) Lens

1) Place lens approximately 1/2 inch aft of light assembly slide forward into position.
2) Place lens retainer in position over aft end of lens.
3) Secure lens retainer and lens with two (2) socket head screws.

4. Ground Recognition Light

When installed, the Ground Recognition Light is located at the top of the rudder assembly. It receives power from the main bus through the 7.5 amp STROBE LIGHTS circuit breaker, and is controlled by the three-position strobe light switch in the overhead switch panel. The Ground Recognition Light is activated individually by placing the strobe light switch in the Fin Strobe position. (See “Figure 5” on page 334013.)

A. Removal

(1) Ensure that the BATT MASTR and STROBE LIGHT switches are in OFF position.
(2) Disengage (pull out) STROBE LIGHTS circuit breaker.
(3) Remove sealant and two (2) screws securing light to the fairing/plate assembly and remove light assembly.
(4) Disconnect electrical leads.

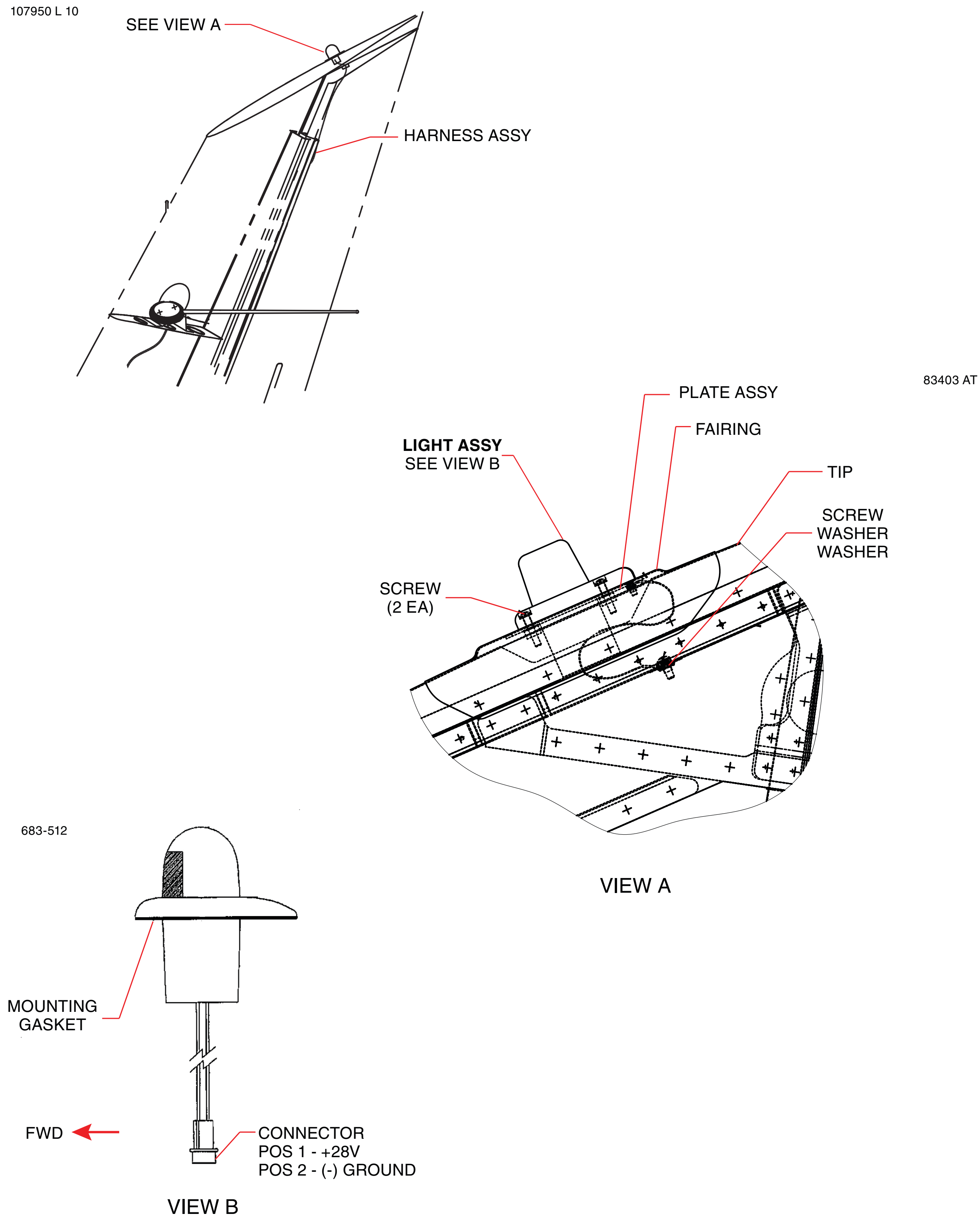
B. Installation

NOTE: Assure clean mating surface.

(1) Reconnect electrical leads and place light into position on rudder fairing/plate assembly.

CAUTION: DO NOT OVER-TIGHTEN SCREWS. AFTER SCREWS ARE FULLY INSTALLED, A SMALL GAP BETWEEN THE LIGHT ASSEMBLY AND THE FAIRING WILL STILL BE PRESENT AND IS ACCEPTABLE.

(2) Secure with two (2) screws.
(3) Engage (push in) STROBE LIGHTS circuit breaker.



Ground Recognition Light
Figure 5

(4) Verify that light is working by selecting BATT MASTR and FIN STROBE light switches ON and checking that the Ground Recognition Light is functioning. Select BATT MASTR and FIN STROBE light switches OFF.

(5) Fillet seal evenly around entire light assembly to fairing mating surface.

(6) Fill screw holes with sealant and smooth to match outer surface of light assembly contour.

(7) Paint as desired.

C. Lamp Replacement

(1) Remove light assembly per Removal, above.

(2) Disassemble light assembly by separating flasher unit from lens and lens retainer.

(3) Pull old lamp from flasher unit and discard.

(4) Insert new lamp into flasher unit.

(5) Assemble lens and lens retainer onto flasher unit and reinstall on airplane per Installation, above.

5. Tail Light

An LED Tail Light is available in S/N's 4636716, 4636720 and up.

If one LED fails the entire light assembly must be replaced. (See "Figure 6" on page 334015.)

A. Removal

(1) Select BATT MASTR and NAV LIGHT switches OFF.

(2) Disconnect the negative (i.e., ground) battery cable.

(3) Remove the screws securing the tail light assembly to the rudder.

(4) Pull the tail light out enough to expose the electrical connector. Capture the wiring harness on the airplane side of the connector.

(5) Disconnect the electrical connector.

(6) Remove the taillight.

B. Installation

(1) Connect the tail light electrical leads to the wiring harness.

(2) Release the electrical harness and feed it back into the airplane until the tail light is in position.

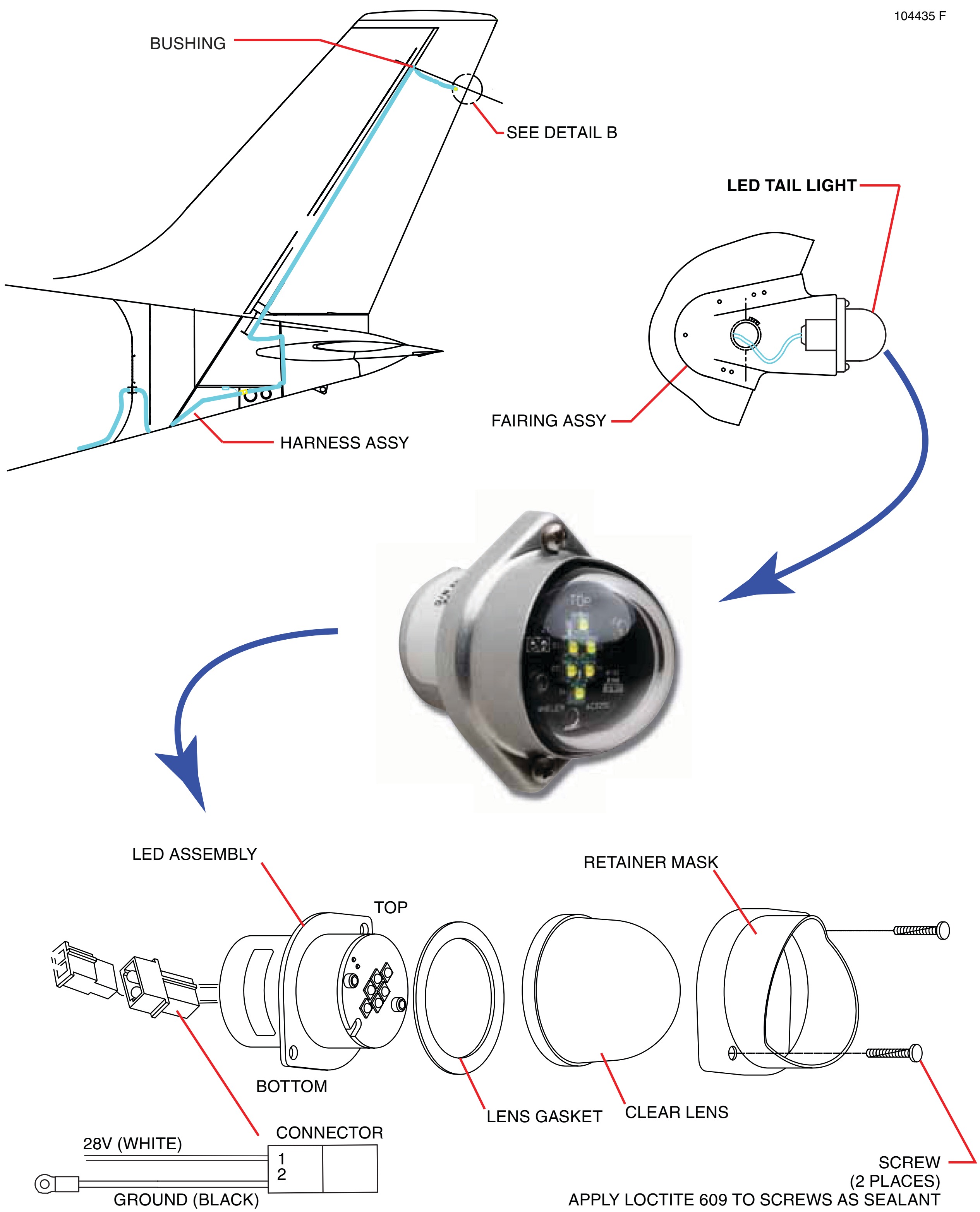
CAUTION : APPLY LOCTITE 609 TO SCREWS TO SEAL AGAINST WATER INTRUSION .

(3) Apply Loctite 609 to screws and secure the tail light to the fairing assembly.

(4) Connect the negative (i.e., ground) battery cable.

(5) Select BATT MASTR and NAV LIGHT switches ON, verify tail light illuminates.

(6) Select BATT MASTR and NAV LIGHT switches OFF, secure airplane.



Tail Light
Figure 6

Effectivity
4636716, 4636720 and up

6. Wing Inspection Light

The wing inspection light is located on the left-hand side of the fuselage at F.S. 129.48. The light is used to help the pilot inspect the wing for any ice formation on the wing leading edge. Two installations have been used: one with a halogen light assembly and another with a LED light assembly. Both are replaced as a unit. (See “Figure 7” on page 334017.)

NOTE: The halogen lamp installation can be replaced with the LED installation at the owner’s descretion.

A. Removal

(1) Remove pilot’s side panel.

(2) From the outside, remove the four (4) screws (one screw is longer than the other three) securing the light assembly to the doubler. The longer screw is from the upper aft position.

(3) Push the light assembly inward.

(4) Disconnect the P/J349 connector.

(5) Remove the light assembly and discard.

B. Installation

(1) Clean any remaining sealant from both sides of the shim and the inboard side of doubler.

(2) Apply Type 2 sealant to both sides of the shim. See 91-10-00, Chart 11, for Type 2 sealant.

(3) Install the shim and new light assembly.

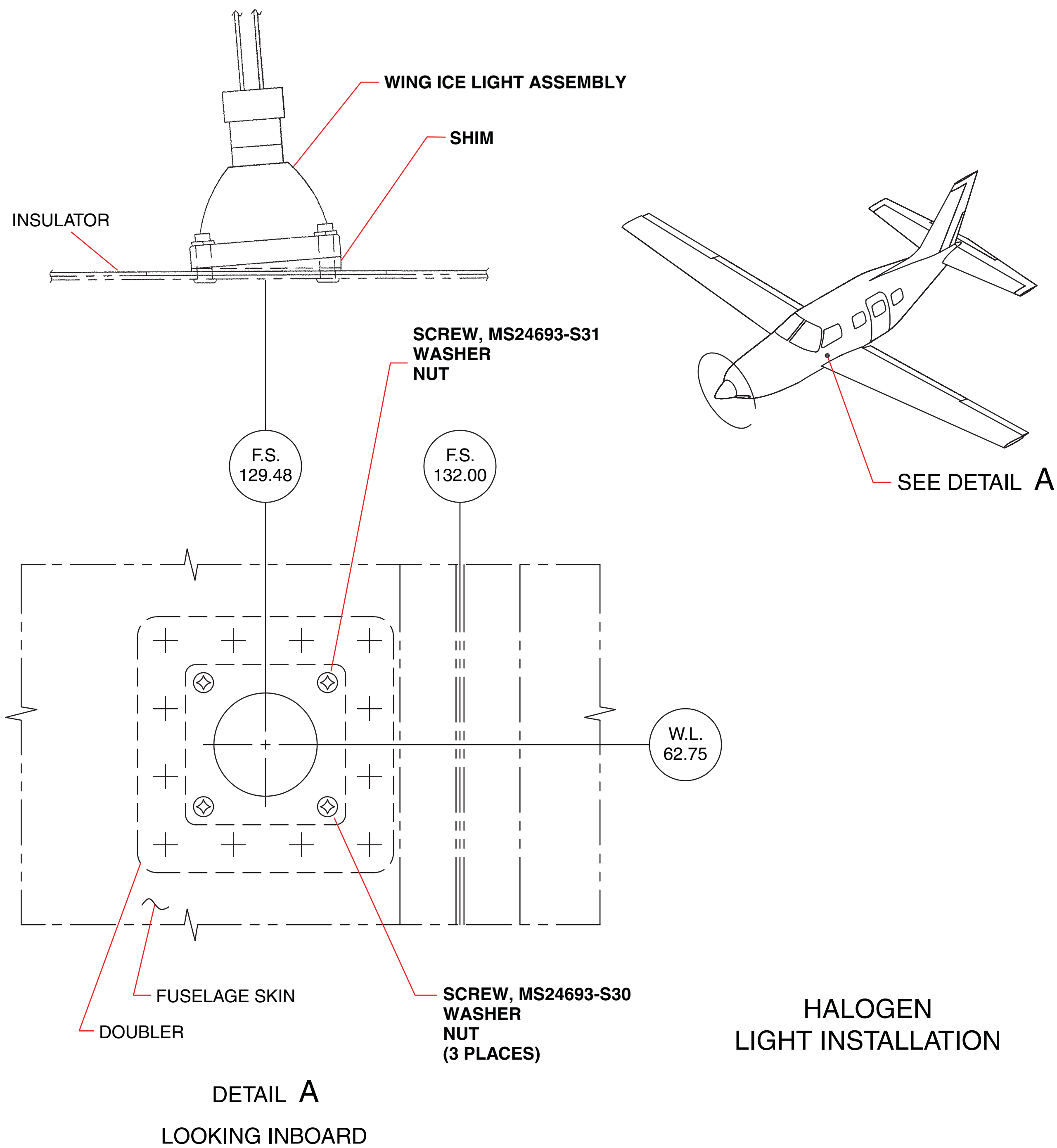
NOTE: The thicker end of the shim should be aft.

(4) Secure with the four (4) screws. Make sure longer screw is inserted into the upper aft position.

(5) Connect the P/J349 connector.

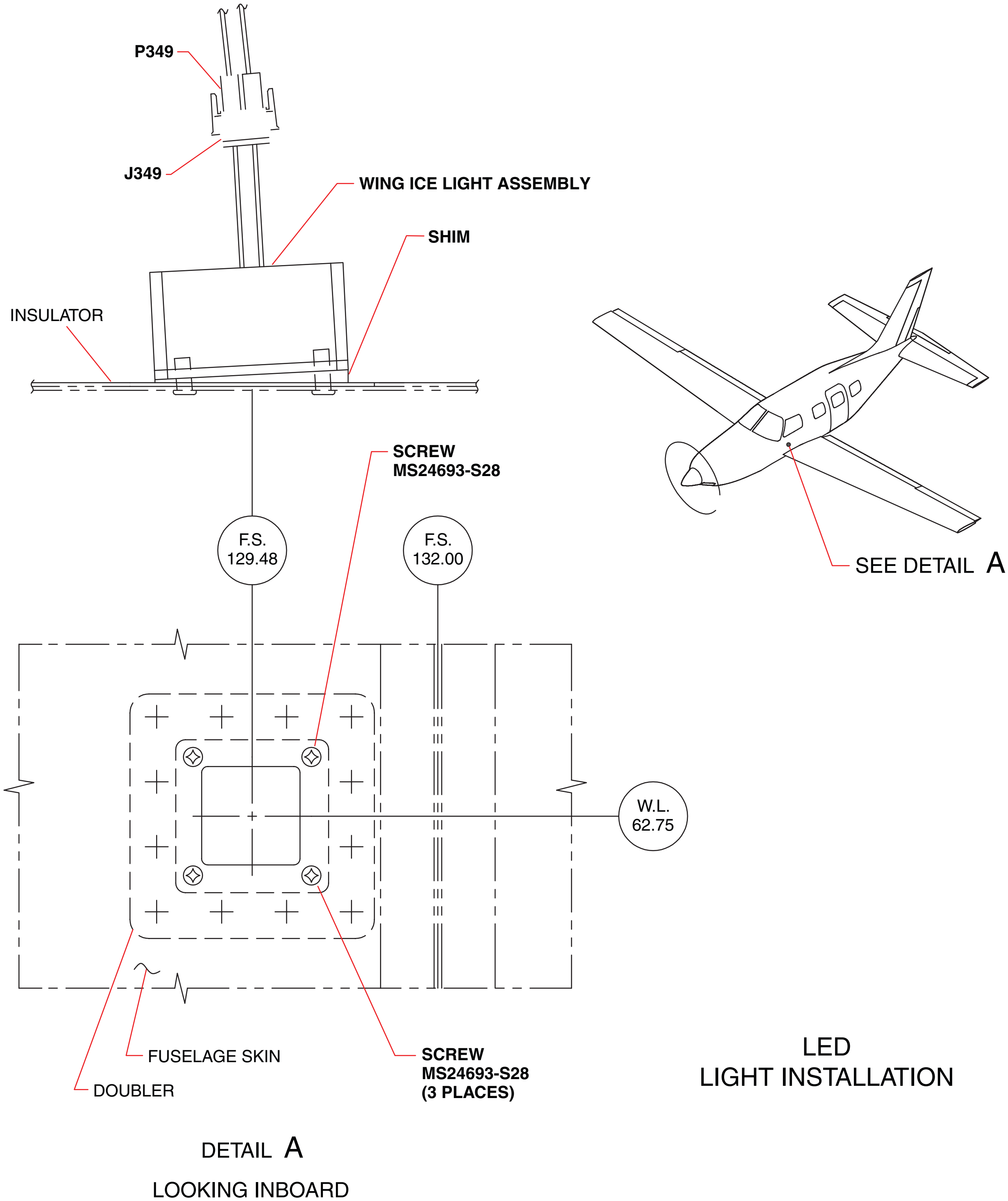
(6) Check operation of light.

(7) Install pilot’s side panel.



Wing Inspection Light
Figure 7 (Sheet 1 of 2)

Effectivity
4636001–4636727



Effectivity
4636728 and up

Wing Inspection Light
Figure 7 (Sheet 2 of 2)

CHAPTER

34

NAVIGATION

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CHAPTER 34

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
34-List of Effective Pages	1	Mar 25/21
	2	Mar 25/21
	3	Mar 25/21
	4	Mar 25/21
	5	Mar 25/21
	6	Mar 25/21
	7	Mar 25/21
	8	Mar 25/21
34-Table of Contents	1	Mar 25/21
	2	Mar 25/21
	3	Mar 25/21
	4	Mar 25/21
	5	Mar 25/21
	6	Mar 25/21
	7	Mar 25/21
	8	Mar 25/21
	9	Mar 25/21
	10	Mar 25/21
	11	Mar 25/21
	12	Mar 25/21
	13	Mar 25/21
	14	Nov 30/17
34-00-00	1	Mar 25/21
	2	Mar 25/21
	3	Mar 25/21
	4	Mar 25/21
34-10-00	1	Mar 25/21
	2	Nov 30/17
	3	Nov 30/17
	4	Nov 30/17
	5	Nov 30/17
	6	Nov 30/17
	7	Nov 30/17
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	28	Mar 25/21
34-20-00	1	Nov 30/17
	2	Mar 25/21
	3	Sep 15/09
	4	Mar 25/21
	5	Nov 30/17
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34-21-00	1	Nov 30/17
	2	Nov 30/17
	3	Nov 30/17
	4	Nov 30/17
	5	Nov 30/17
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	11	Nov 30/17
	12	Nov 30/17
	13	Nov 30/17
	14	Nov 30/17
	15	Nov 30/17
	16	Nov 30/17

CHAPTER 34

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE
34-22-00	1	Nov 30/17
	2	Nov 30/17
	3	Nov 30/17
	4	Nov 30/17
	5	Nov 30/17
	6	Nov 30/17
	7	Nov 30/17
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34-23-00	1	Nov 30/17
	2	Nov 30/17
34-24-00	1	Mar 25/21
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	27	Mar 25/21
	28	Mar 25/21
	29	Mar 25/21
	30	Mar 25/21

CHAPTER 34

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
34-25-01	1	Nov 30/17		47	Mar 25/21
	2	Nov 30/17		48	Mar 25/21
	3	Nov 30/17		49	Mar 25/21
	4	Nov 30/17		60	Nov 30/17
	5	Nov 30/17		61	Nov 30/17
	6	Nov 30/17		62	Nov 30/17
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	40	Nov 30/17			
	41	Nov 30/17			
	42	Nov 30/17			
	43	Nov 30/17			
	44	Nov 30/17			
	45	Mar 25/21			
	46	Mar 25/21			

CHAPTER 34

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
34-25-01 (cont.)	93	Nov 30/17		139	Mar 25/21
	94	Nov 30/17		140	Nov 30/17
	95	Nov 30/17		141	Nov 30/17
	96	Nov 30/17		142	Nov 30/17
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	105	Nov 30/17		151	Nov 30/17
	106	Nov 30/17		152	Mar 25/21
	107	Nov 30/17		153	Nov 30/17
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	131	Nov 30/17		177	Mar 25/21
	132	Nov 30/17		178	Mar 25/21
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	134	Nov 30/17		180	Nov 30/17
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	136	Nov 30/17		182	Nov 30/17
	137	Nov 30/17		183	Nov 30/17
	138	Nov 30/17		184	Nov 30/17

CHAPTER 34

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
34-25-01 (cont.)	185	Nov 30/17		231	Nov 30/17
	186	Nov 30/17		232	Nov 30/17
	187	Nov 30/17		233	Nov 30/17
	188	Nov 30/17		234	Nov 30/17
	189	Nov 30/17		235	Nov 30/17
	190	Nov 30/17		236	Nov 30/17
	191	Nov 30/17		237	Nov 30/17
	192	Nov 30/17		238	Nov 30/17
	193	Nov 30/17		239	Nov 30/17
	194	Nov 30/17		240	Nov 30/17
	195	Nov 30/17		241	Nov 30/17
	196	Nov 30/17		242	Nov 30/17
	197	Nov 30/17		243	Nov 30/17
	198	Nov 30/17		244	Nov 30/17
	199	Nov 30/17		245	Nov 30/17
	200	Nov 30/17		246	Nov 30/17
	201	Nov 30/17		247	Nov 30/17
	202	Nov 30/17		248	Nov 30/17
	203	Nov 30/17		249	Nov 30/17
	204	Nov 30/17		250	Nov 30/17
	205	Nov 30/17		251	Nov 30/17
	206	Nov 30/17		252	Nov 30/17
	207	Nov 30/17		253	Nov 30/17
	208	Nov 30/17		254	Nov 30/17
	209	Nov 30/17		255	Nov 30/17
	210	Nov 30/17		256	Nov 30/17
	211	Nov 30/17			
	212	Nov 30/17	34-25-02	1	Mar 25/21
	213	Nov 30/17		2	Nov 30/17
	214	Nov 30/17		3	Nov 30/17
	215	Nov 30/17		4	Nov 30/17
	216	Nov 30/17		5	Mar 25/21
	217	Nov 30/17		6	Nov 30/17
	218	Nov 30/17		7	Mar 25/21
	219	Nov 30/17		8	Nov 30/17
	220	Nov 30/17		9	Nov 30/17
	221	Nov 30/17		10	Nov 30/17
	222	Nov 30/17		11	Nov 30/17
	223	Nov 30/17		12	Nov 30/17
	224	Nov 30/17		13	Nov 30/17
	225	Nov 30/17		14	Nov 30/17
	226	Nov 30/17		15	Nov 30/17
	227	Nov 30/17		16	Nov 30/17
	228	Nov 30/17		17	Nov 30/17
	229	Nov 30/17		18	Mar 25/21
	230	Nov 30/17		19	Nov 30/17

CHAPTER 34

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
34-25-02 (cont.)	20	Nov 30/17		66	Mar 25/21
	21	Nov 30/17		67	Mar 25/21
	22	Mar 25/21		68	Nov 30/17
	23	Nov 30/17		69	Nov 30/17
	24	Nov 30/17		70	Nov 30/17
	25	Nov 30/17		71	Nov 30/17
	26	Nov 30/17		72	Nov 30/17
	27	Nov 30/17		73	Nov 30/17
	28	Nov 30/17		74	Nov 30/17
	29	Nov 30/17		75	Nov 30/17
	30	Nov 30/17		76	Nov 30/17
	31	Nov 30/17		77	Nov 30/17
	32	Nov 30/17		78	Nov 30/17
	33	Nov 30/17		79	Nov 30/17
	34	Mar 25/21		80	Nov 30/17
	35	Nov 30/17		81	Nov 30/17
	36	Nov 30/17		82	Nov 30/17
	37	Mar 25/21		83	Nov 30/17
	38	Mar 25/21		84	Nov 30/17
	39	Nov 30/17		85	Mar 25/21
	40	Mar 25/21		86	Mar 25/21
	41	Mar 25/21		87	Mar 25/21
	42	Nov 30/17		88	Nov 30/17
	43	Mar 25/21		89	Mar 25/21
	44	Nov 30/17		90	Mar 25/21
	45	Nov 30/17		91	Mar 25/21
	46	Nov 30/17		92	Mar 25/21
	47	Nov 30/17		93	Mar 25/21
	48	Mar 25/21		94	Mar 25/21
	49	Mar 25/21		95	Mar 25/21
	50	Mar 25/21		96	Nov 30/17
	51	Mar 25/21		97	Nov 30/17
	52	Mar 25/21		98	Nov 30/17
	53	Mar 25/21		99	Nov 30/17
	54	Mar 25/21		100	Nov 30/17
	55	Mar 25/21		101	Mar 25/21
	64	Nov 30/17		102	Mar 25/21
	65	Mar 25/21		103	Nov 30/17
				104	Nov 30/17
				105	Nov 30/17
				106	Nov 30/17
				107	Mar 25/21
				108	Nov 30/17
				109	Mar 25/21
				110	Mar 25/21
				111	Mar 25/21

CHAPTER 34

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
34-25-02 (cont.)	112	Mar 25/21		158	Mar 25/21
	113	Mar 25/21		159	Mar 25/21
	114	Mar 25/21		160	Mar 25/21
	115	Mar 25/21		161	Mar 25/21
	116	Mar 25/21		162	Mar 25/21
	117	Mar 25/21		163	Nov 30/17
	118	Mar 25/21		164	Nov 30/17
	119	Nov 30/17		165	Mar 25/21
	120	Nov 30/17		166	Mar 25/21
	121	Mar 25/21		167	Nov 30/17
	122	Mar 25/21		168	Nov 30/17
	123	Mar 25/21		169	Nov 30/17
	124	Mar 25/21		170	Nov 30/17
	125	Mar 25/21		171	Nov 30/17
	126	Nov 30/17		172	Nov 30/17
	127	Mar 25/21		173	Nov 30/17
	128	Nov 30/17		174	Mar 25/21
	129	Nov 30/17		175	Mar 25/21
	130	Mar 25/21		176	Mar 25/21
	131	Nov 30/17		177	Mar 25/21
	132	Nov 30/17		178	Mar 25/21
	133	Nov 30/17		179	Mar 25/21
	134	Nov 30/17		180	Nov 30/17
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	136	Nov 30/17		182	Nov 30/17
	137	Nov 30/17		183	Nov 30/17
	138	Mar 25/21		184	Mar 25/21
.	139	Mar 25/21		185	Nov 30/17
	140	Nov 30/17		186	Nov 30/17
	141	Mar 25/21		187	Nov 30/17
	142	Nov 30/17		188	Nov 30/17
	143	Nov 30/17		189	Nov 30/17
	144	Nov 30/17		190	Mar 25/21
	145	Nov 30/17		191	Nov 30/17
	146	Nov 30/17		192	Nov 30/17
	147	Nov 30/17		193	Mar 25/21
	148	Nov 30/17		194	Nov 30/17
	149	Nov 30/17		195	Nov 30/17
	150	Mar 25/21		196	Nov 30/17
	151	Mar 25/21		197	Nov 30/17
	152	Nov 30/17		198	Nov 30/17
	153	Mar 25/21		199	Mar 25/21
	154	Nov 30/17		200	Mar 25/21
	155	Nov 30/17		201	Mar 25/21
	156	Nov 30/17		202	Nov 30/17
	157	Mar 25/21		203	Mar 25/21

CHAPTER 34

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE
34-25-02 (cont.)	204	Nov 30/17
	205	Nov 30/17
	206	Mar 25/21
	207	Mar 25/21
	208	Mar 25/21
	209	Nov 30/17
	210	Nov 30/17
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	230	Mar 25/21
	231	Mar 25/21
	232	Mar 25/21
	233	Mar 25/21
	234	Mar 25/21
	235	Mar 25/21
	236	Mar 25/21
34-30-00	1	Mar 25/21
	2	Mar 25/21
34-40-00	1	Mar 25/21
	2	Nov 30/17
	3	Sep 15/09
	4	Sep 15/09
	5	Sep 15/09
	6	Sep 15/09
	7	Sep 15/09
	8	Sep 15/09
34-50-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Sep 15/09
	5	Sep 15/09
	6	Jan 15/13
	7	Mar 25/21
	8	Mar 25/21
	9	Mar 25/21
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	11	Mar 25/21
	12	Mar 25/21
	13	Mar 25/21
	14	Mar 25/21
	15	Mar 25/21
	16	Mar 25/21
	17	Mar 25/21
	18	Mar 25/21
	19	Nov 30/17
	20	Nov 30/17
	21	Mar 25/21
	22	Mar 25/21
	23	Mar 25/21
	24	Mar 25/21
	25	Mar 25/21
	26	Mar 25/21

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

CHAPTER 34 - NAVIGATION

TABLE OF CONTENTS (continued)

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GENERAL

1. 1995 thru 2005

A. Standard

Traditional 3.25 Inch Flight Instruments were installed as standard equipment in S/N's 4636001–4636374. These instruments are face mounted, see 39-10-00 for removal and installation.

B. Optional

In S/N's 4636299, 4636333–4636374 only, the Meggitt Avionics next Generation Integrated Cockpit (MAGIC) Electronic Flight Instrument System (EFIS) was available as an option on the pilot's side only. When installed, it's the primary flight and navigation instrumentation. The MAGIC EFIS consists of three primary components:

(1) Primary Flight Display (PFD) [i.e. - Electronic Attitude Direction Indicator (EADI)],

(2) Navigation Display (NAV) [i.e. - Electronic Horizontal Situation Indicator (EHSI)], and

(3) Air Data & Attitude Heading Reference System (ADAHRS) [i.e. - Attitude Heading Reference System (AHRS)].

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-21-00.

2. 2006 thru 2010

The Avidyne FlightMax Entegra Integrated Flight Display System (IFDS) was installed as standard equipment in S/N's 4636375 thru 4636459, 4636461 thru 4636462, and 4636481; and 4692001 thru 4692133, 4692141, 4692149 and 4692153.

This system uses three (PA-46-350P) or two (PA-46R-350T) large 10.4-inch diagonal, high-resolution, sunlight-readable full color displays (PFD and MFD), to provide standard flight instrumentation including attitude direction indicator (EADI), horizontal situation indicator (EHSI), altitude, airspeed, vertical speed, moving map, weather, terrain and traffic data, as well as all engine instrumentation. Standby flight instruments (i.e. - airspeed, electric attitude indicator, and altimeter) provide redundancy. The PA-46-350P uses two-inch standby instruments and the PA-46R-350T uses three inch standby instruments. (See 34-22-00, "Figure 1" on page 34208.)

The EFIS installation consists of the following components: pilot and co-pilot Primary Flight Displays (PFDs); a single Multifunction Display (MFD); Data Acquisition Unit (DAU), and associated sensors; and two Magnetometer/OAT Sensor Assemblies.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-22-00.

See 77-40-00 for additional info on the Data Acquisition Unit (DAU).

3. 2010 thru 2017

The Garmin G1000 Integrated Avionics System (IAS) was installed as standard equipment in PA-46-350P S/N's 4636460, 4636463–4636715, 4636717–4636719; and PA-46R-350T S/N's 4692134–4692140, 4692142–4692148, 4692150–4692152, 4692154–4692214.

The Garmin G1000 system has a three panel configuration with two GDU 1040 10.4-inch LCD panels as Primary Flight Display (PFDs), and one GDU 1500 15-inch or GDU 1240A 12-inch LCD panel as the Multi-Function Display (MFD). Functions provided by the system present flight instrumentation, position, navigation, communication, and identification information. The system consists of the following Line Replaceable Units (LRUs):

- GIA 63W Integrated Avionics Unit (2 ea.)
- GDC 74A Air Data Computer (ADC) (2 ea.)
- GEA 71 Engine/Airframe Unit
- GRS 77 Attitude and Heading Reference System (AHRS) (2 ea.)
- GMU 44 Magnetometer (2 ea.)
- GMA 347 Audio Panel with Integrated Marker Beacon Receiver, or
- GMA 350 Audio Panel with Integrated Marker Beacon Receiver(1)
- GTX 33/33D/33 ES/33D ES Mode S Transponder (1 standard, 2 optional)
- GDL 69A/69eA Satellite Data Link Receiver (optional)
- GWX 68 Weather Radar (optional)
- GSR 56 Iridium Satellite Receiver (optional) (1)
- GTS 825 Traffic Advisory System (TAS) (optional) (1)
- GCU 476 MFD/PFD Control Unit
- GMC 710 AFCS Control Unit
- GTP 59 Outside Air Temperature (OAT) Probe (2 ea.)
- GSA 8X Servo Actuators (4 ea.)
- GSM 8X Servo Gearboxes (4 ea.)
- GAE 43 Cabin Altitude Data System

Standby altitude and airspeed indicators and a magnetic compass are also installed as secondary, independent sources of altitude, airspeed, and (non-stabilized) heading information. An electric attitude indicator is installed to provide an independent source for attitude information.

In S/N's 4636633, 4636652– 4636715, 4636717–4636719, the individual standby instruments are replaced by an Aspen Avionics Evolution Backup Display (EBD).

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Line maintenance of these systems is limited to basic troubleshooting and component replacement. See "Aspen Avionics Evolution Backup Display (EBD)" on page 34241 and "Integrated Avionics System (IAS) - Garmin G1000" on page 3425011.

(1) 4636633, 4636652 and up.

4. 2017 thru 2020

The Garmin Intergrated Avionics System (IAS) is installed as standard equipment in S/N's 4636720–4636775. The Garmin G1000 NXi system has a three panel configuration with two GDU 105X LCD panels as Primary Flight Display (PFDs), and one GDU 1250A LCD panel as the Multi-Function Display (MFD). Functions provided by the system present flight instrumentation, position, navigation, communication, and identification information. The system consists of the following Line Replaceable Units (LRUs):

- GIA 64 Integrated Avionics Unit (2 ea.) (S/N's 4636716, 4636720–4636759)

  or

- GIA 64W Integrated Avionics Unit (2 ea.) (S/N's 4636760–4636775)
- GDC 72 Air Data Computer (ADC) (2 ea.)
- GEA 71B Engine/Airframe Unit
- GRS 79 Attitude and Heading Reference System (AHRS) (2 ea.)
- GMU 44 Magnetometer (2 ea.)
- GMA 350c Audio Panel
- GTX 335R Transponder (1 ea. standard 1 ea. optional)
- GTX 33D ES Mode S Transponder (2 ea.) (optional - when diversonary transponder required)
- GTX 345R Transponder (2 ea.) (optional)
- GDL 69eA Sirius/XM Weather/Music (optional)
- GWX 68 Weather Radar (optional)
- GSR 56 Iridium Satellite Receiver (optional)
- GTS 825 Traffic Advisory System (TAS) (optional)
- GCU 47X MFD/PFD Control Unit
- GMC 710 AFCS Control Unit
- GTP 59 Outside Air Temperature (OAT) Probe (2 ea.)
- GSA 8X Servo Actuators (4 ea.)
- GSM 8X Servo Gearboxes (4 ea.)
- WX-500 Stromscope (optional)
- FS 510 Bluetooth® and Wi-Fi Connectivity (optional)
- GAE 43 Cabin Altitiude Data System

An Aspen Avionics Evolution Backup Display (EBD) is also installed as a secondary, independent source of attitude, altitude, airspeed, vertical speed and heading information.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Line maintenance of these systems is limited to basic troubleshooting and component replacement. See "Aspen Avionics Evolution Backup Display (EBD)" on page 34241 and "Integrated Avionics System (IAS) - Garmin G1000 NXi" on page 3425021.

5. 2021 and up

The Garmin G1000 NXi Intergrated Avionics System (IAS) is installed as standard equipment in S/N's 4636776 and up.

The Garmin G1000 NXi system has a three panel configuration with two GDU 105X LCD panels as Primary Flight Display (PFDs), and one GDU 1250A LCD panel as the Multi-Function Display (MFD). Functions provided by the system present flight instrumentation, position, navigation, communication, and identification information. The system consists of the following Line Replaceable Units (LRUs):

- GIA 64W Integrated Avionics Unit (2 ea.)
- GDC 72 Air Data Computer (ADC) (2 ea.)
- GEA 71B Engine/Airframe Unit
- GRS 79 Attitude and Heading Reference System (AHRS) (2 ea.)
- GMU 44B Magnetometer (2 ea.)
- GMA 350c Audio Panel
- GTX 345R Transponder (1 ea. standard)
- GTX 345DR Transponder (1 ea. optional)
- GTX 335R Transponder (1 ea. optional)
- GDL 69eA Sirius/XM Weather/Music (optional)
- GWX 75 Weather Radar (optional)
- GSR 56 Iridium Satellite Receiver (optional)
- GTS 825 Traffic Advisory System (TAS) (optional)
- GCU 47X MFD/PFD Control Unit
- GMC 710 AFCS Control Unit
- GTP 59 Outside Air Temperature (OAT) Probe (2 ea.)
- GSA 8X Servo Actuators (4 ea.)
- GSM 8X Servo Gearboxes (4 ea.)
- WX-500 Stromscope (optional)
- FS 510 Bluetooth® and Wi-Fi Connectivity (optional)
- GAE 43 Cabin Altitiude Data System

A Garmin GI 275 Standby Display is also installed as a secondary, independent source of attitude, altitude, airspeed, vertical speed and heading information.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Line maintenance of these systems is limited to basic troubleshooting and component replacement. See "Garmin GI 275 Standby Instrument" on page 3424011 and "Integrated Avionics System (IAS) - Garmin G1000 NXi" on page 3425021.

FLIGHT ENVIRONMENT DATA

The instrument air system consists of pitot air and static air sources.

In conventional gauge-equipped, Meggitt EFIS-equipped, or Avidyne Entegra-equipped aircraft, the system supplies both pitot and static air pressure for the airspeed indicator, and static pressure for the altimeter, vertical speed indicator, and the triple indicator (cabin pressure differential, vertical speed, and altitude). When installed, pitot and static air pressure is also provided for either the Meggitt EFIS or Avidyne Entegra IFDS Air Data and Attitude Heading Reference System (ADAHRS).

In Garmin G1000/G1000 NXi equipped aircraft, the system supplies both pitot and static air pressure for the Air Data Computers (GDC 74A/72), standby airspeed indicator (early), Aspen Avionics EBD (later), or Garmin GI 275 (2021 and up); as well as static pressure for the standby altimeter (early). Instruments addressed in this section are face mounted. (See 39-10-00, Face Mounted Instruments - Removal and Installation.)

1. Pitot and Static Systems

   A. Description and Operation

      See "Figure 1" on page 34102.

      The pitot air system consists of pitot mast (located on the underside of the left wing for all installations) with its related plumbing. Avidyne dual PFD and Garmin G1000 installations include pitot mast locations on the underside of the left and right wings. In conventional gauge-equipped aircraft thru Avidyne Entegra-equipped aircraft, impact air pressure entering the pitot tube is transmitted from the pitot inlet, through hose and tubing, routed through the wing to the airspeed indicator(s), located on the instrument panel; and, if installed, to the ADAHRS unit located on the aft equipment shelf (Meggitt) or in each PFD (Avidyne). In Garmin G1000/G1000 NXi equipped aircraft, the impact air pressure is carried to the Air Data Computers and to the standby airspeed gauge (early) or Aspen Avionics EBD (later) on the instrument panel.

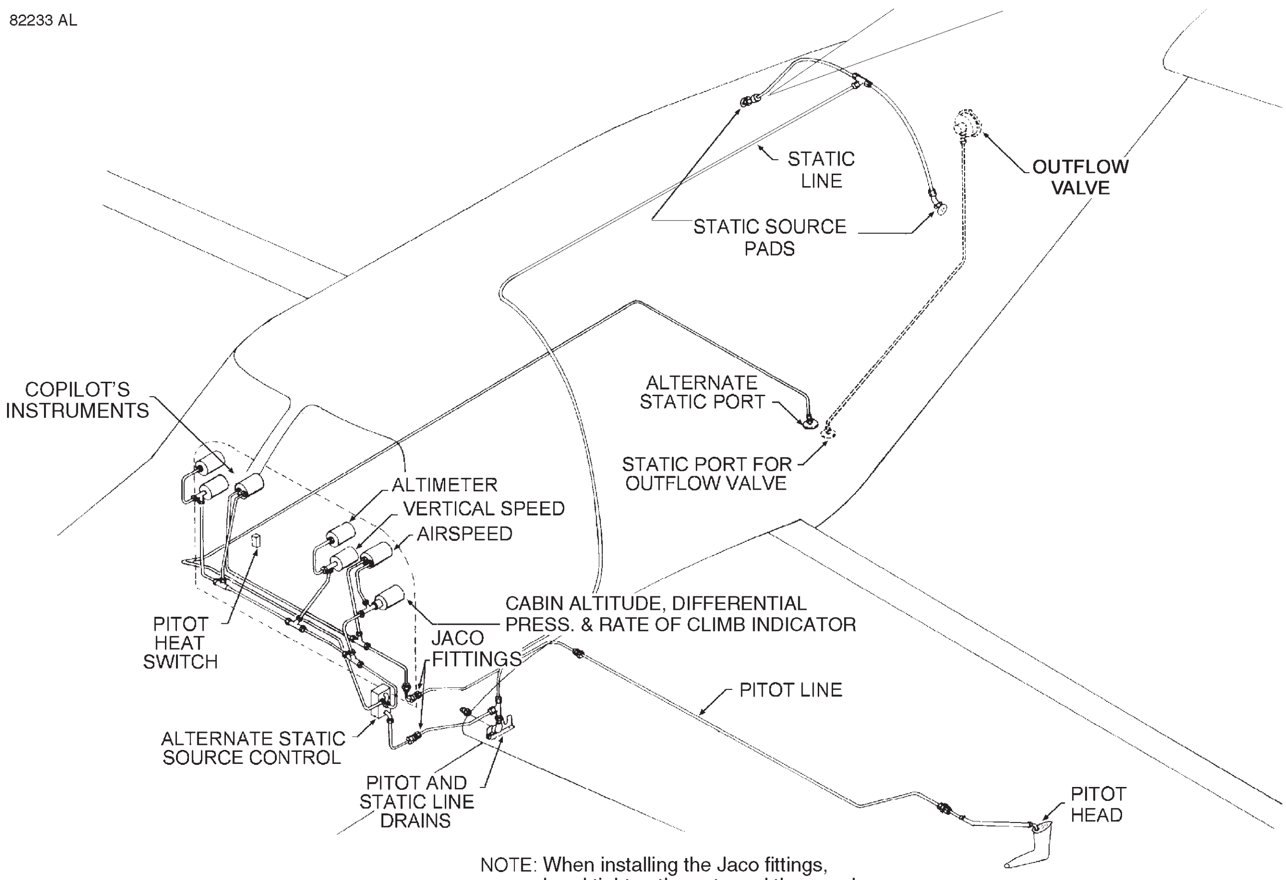
      In conventional gauge-equipped, Meggitt EFIS-equipped, or Avidyne Entegra-equipped aircraft, the static air source consists of either two static ports, one on each side of the fuselage (S/N's 4636001 thru 4636374 and S/N's 4692001 and up) or four static ports, two on each side of the fuselage (S/N's 4636375 and up) aft of F.S. 265.80. The static air system is connected to the airspeed indicator(s), altimeter(s), vertical speed indicator and the triple indicator (cabin pressure differential, vertical speed, and altitude), located on the instrument panel; and, if installed, to the ADAHRS unit located on the aft equipment shelf (Meggitt) or in each PFD (Avidyne); by means of hose and tubing routed through the fuselage.

      In Garmin G1000/G1000 NXi equipped aircraft, the static pressure for the Air Data Computers, standby altimeter (early), standby airspeed indicator (early), Aspen Avionics EBD (later), or Garmin GI 275 (2021 and up), is sensed by four static ports, two (pilot and copilot) on each side of the fuselage aft of F.S. 265.80.

      In the event the primary static source becomes blocked, the system incorporates an external alternate static port located on the bottom of the fuselage aft of F.S. 241.50. A shutoff valve, located below the instrument panel to the left of the pilot, opens or closes the alternate static port as required. A placard giving instructions for use of alternate static pressure, is located on the instrument panel or pilot's window frame. Static pressure for the pressurization system outflow valve (PA-46-350P only) is sensed by another static pad at F.S. 243.38.

      WARNING: DO NOT ATTEMPT TO DRAIN STATIC SYSTEM DURING PRESSURIZED FLIGHT.

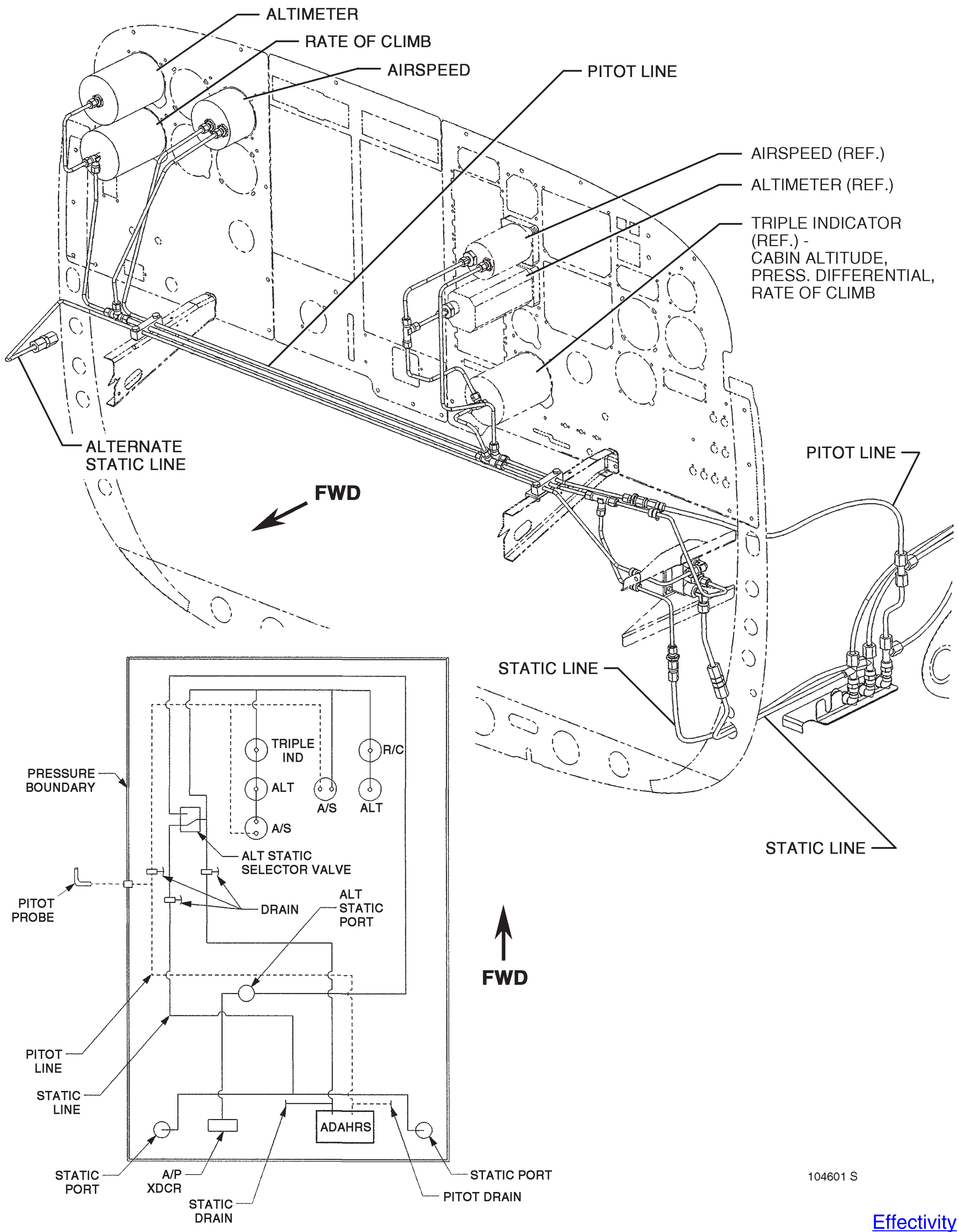
      If one or more of the pitot static instruments malfunction, the system should be checked for dirt, leaks or moisture. A partially or completely blocked pitot head will give erratic, erroneous, or zero reading on the instruments. Pitot and static lines can be drained through the drain valve located on the left lower side of the fuselage interior.



NOTE: When installing the Jaco fittings, hand tighten the nuts and then apply two (2) more turns.

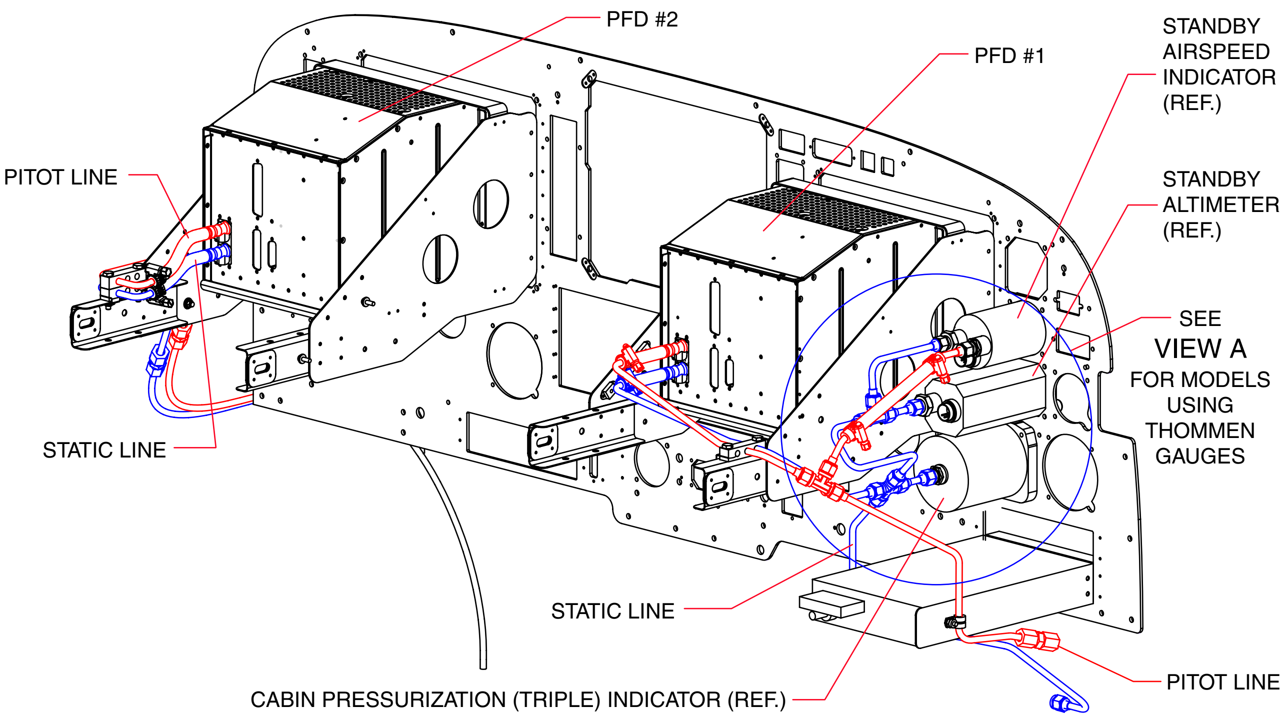
Effectivity
4636001 thru 4636374

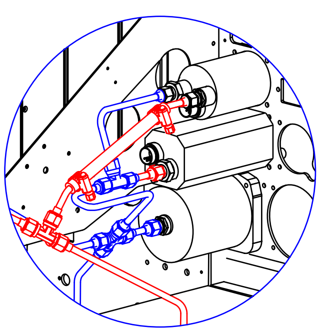
Pitot-Static System Installation
Figure 1 (Sheet 1 of 12)



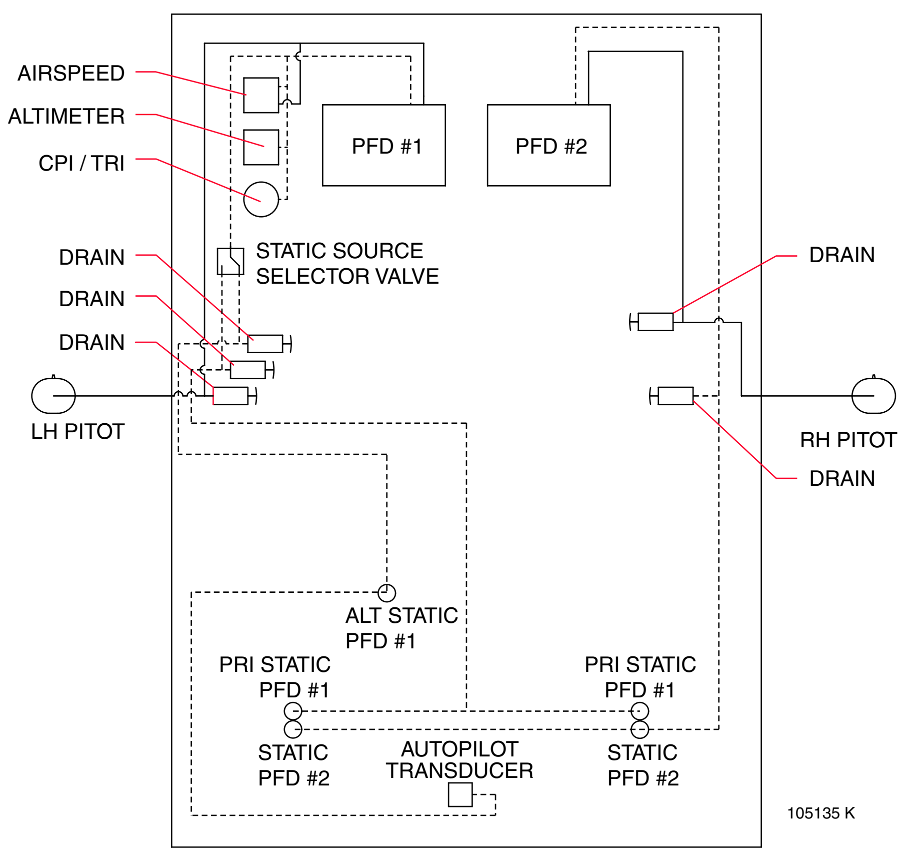
Pitot-Static System Installation
Figure 1 (Sheet 2 of 12)

Effectivity
with Meggitt EFIS Option in
4636299, 4636333–4636374



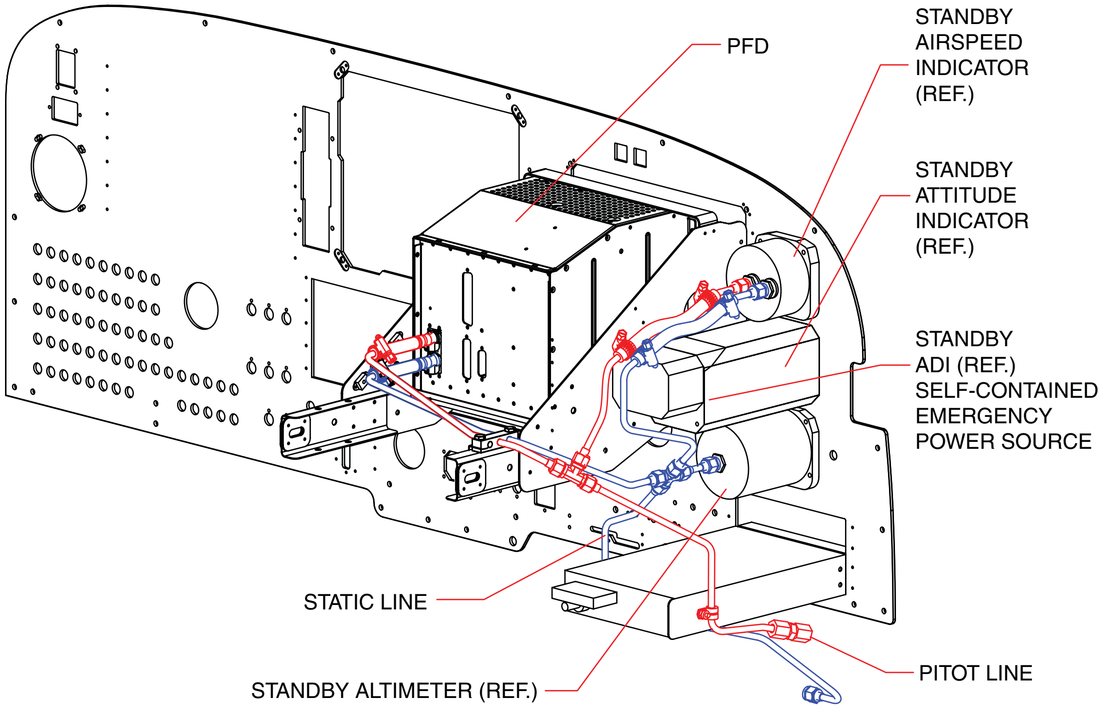


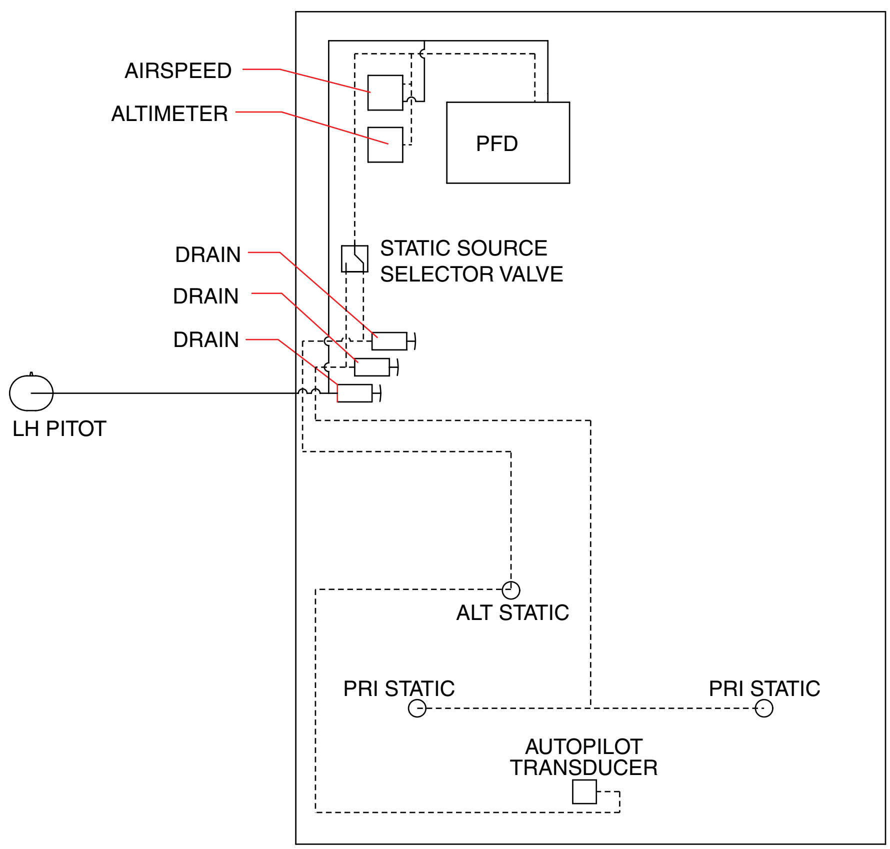
VIEW A
LOOKING AFT
FOR S/Ns 4636457 AND UP
USING THOMMEN GAUGES



Effectivity
4636375–4636459
4636461–4636462, 4636481

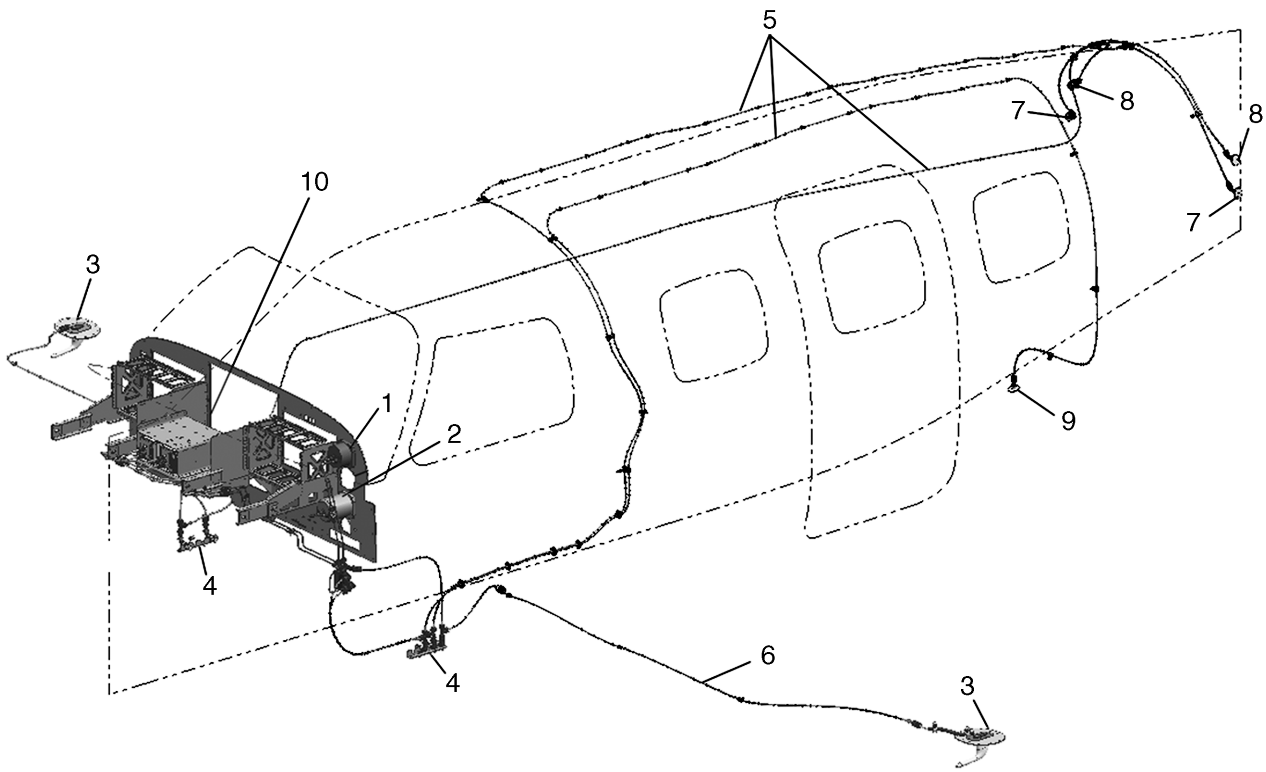
Pitot-Static System Installation
Figure 1 (Sheet 3 of 12)





Pitot-Static System Installation
Figure 1 (Sheet 4 of 12)

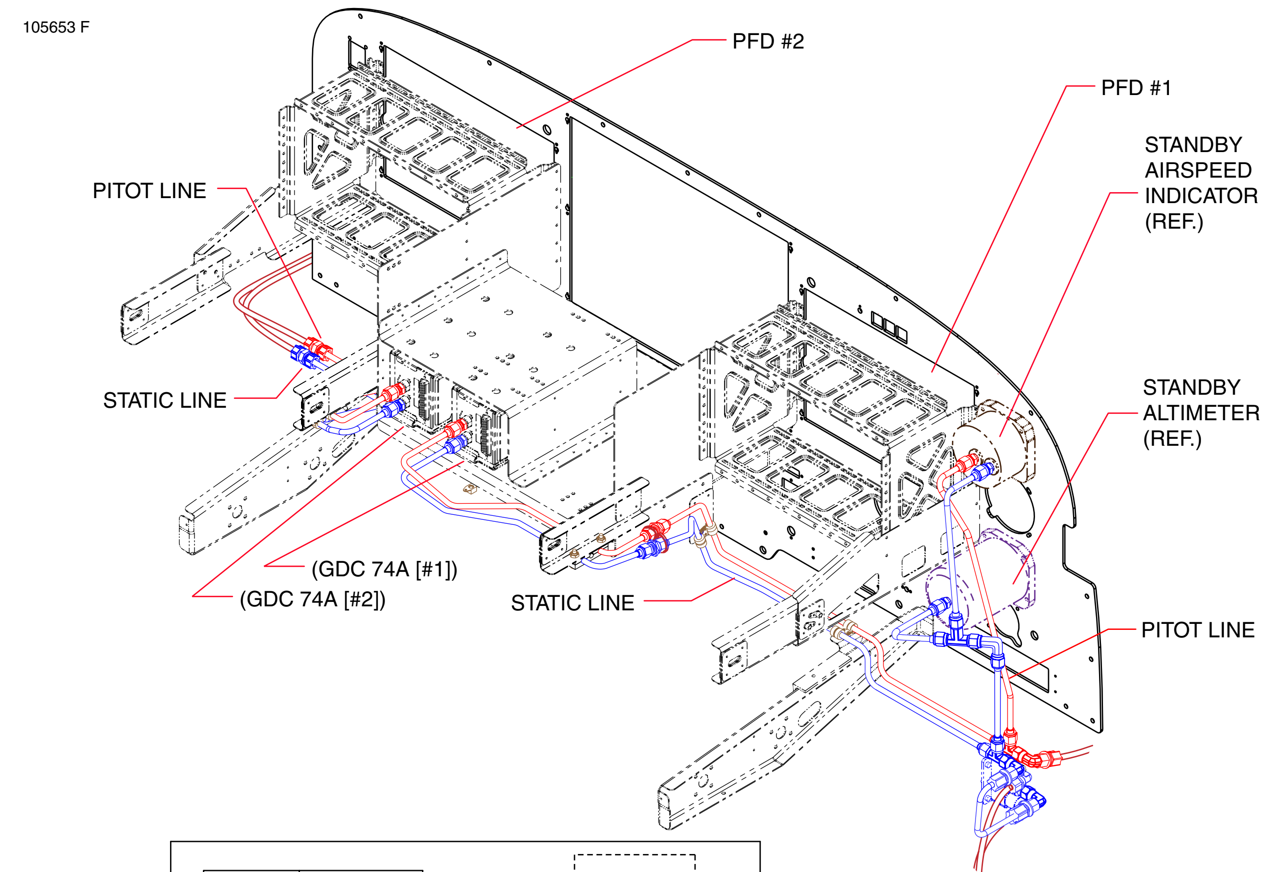
Effectivity
4692001–4692133, 4692141
4692149 and 4692153

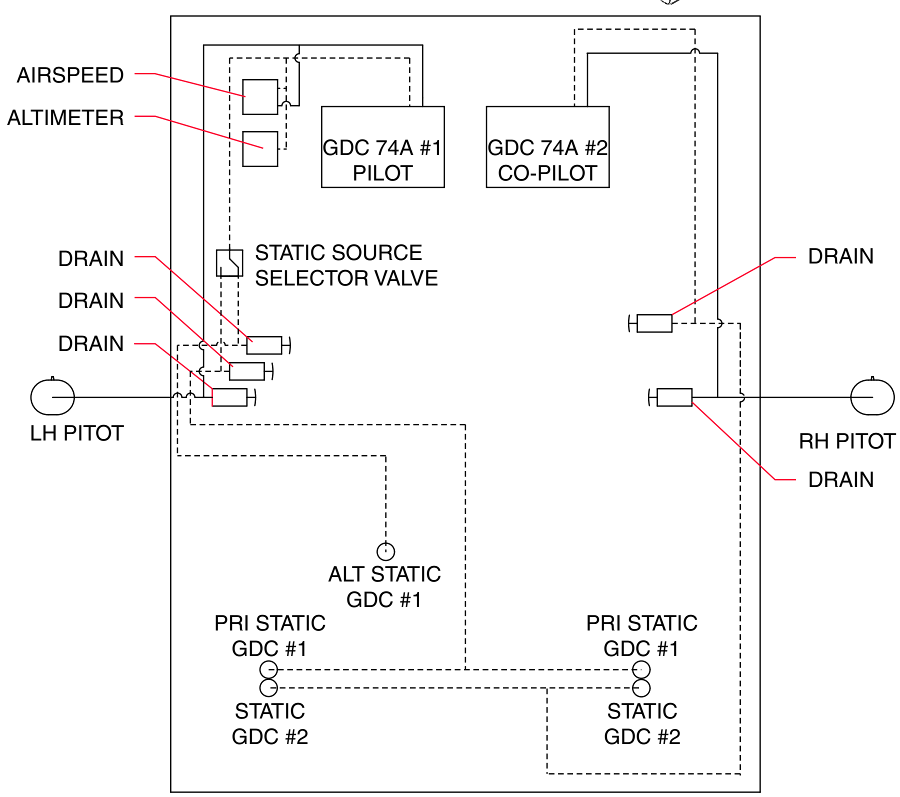


1. STANDBY AIRSPEED
2. STANDBY ALTIMETER
3. PITOT HEAD
4. PITOT/STATIC LINES DRAIN
5. STATIC LINE
6. PITOT LINE
7. PILOT STATIC PORT
8. CO-PILOT STATIC PORT
9. ALTERNATE STATIC PORT
10. PITOT HEAT SWITCH

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153

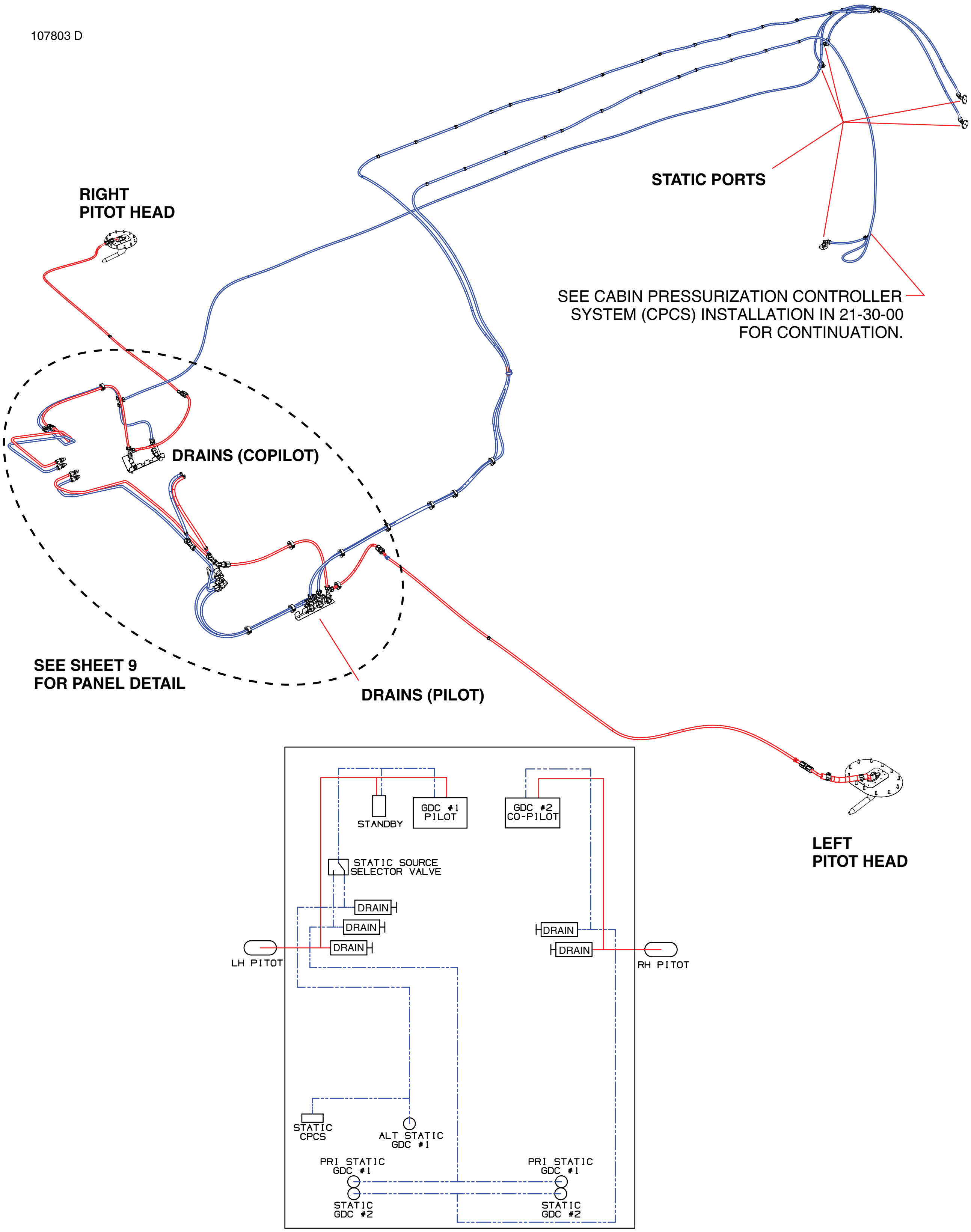
Pitot-Static System Installation
Figure 1 (Sheet 5 of 12)





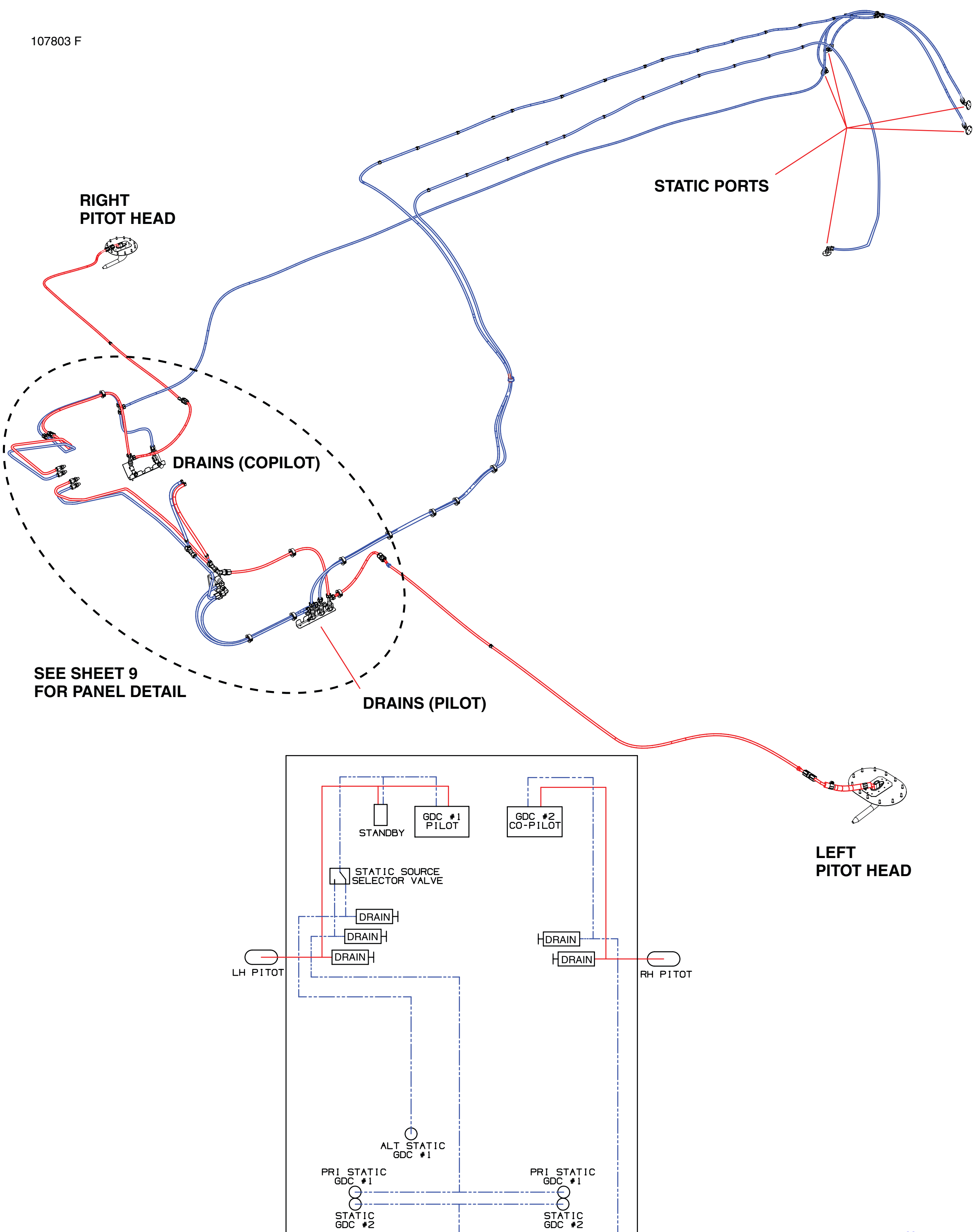
Pitot-Static System Installation
Figure 1 (Sheet 6 of 12)

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153



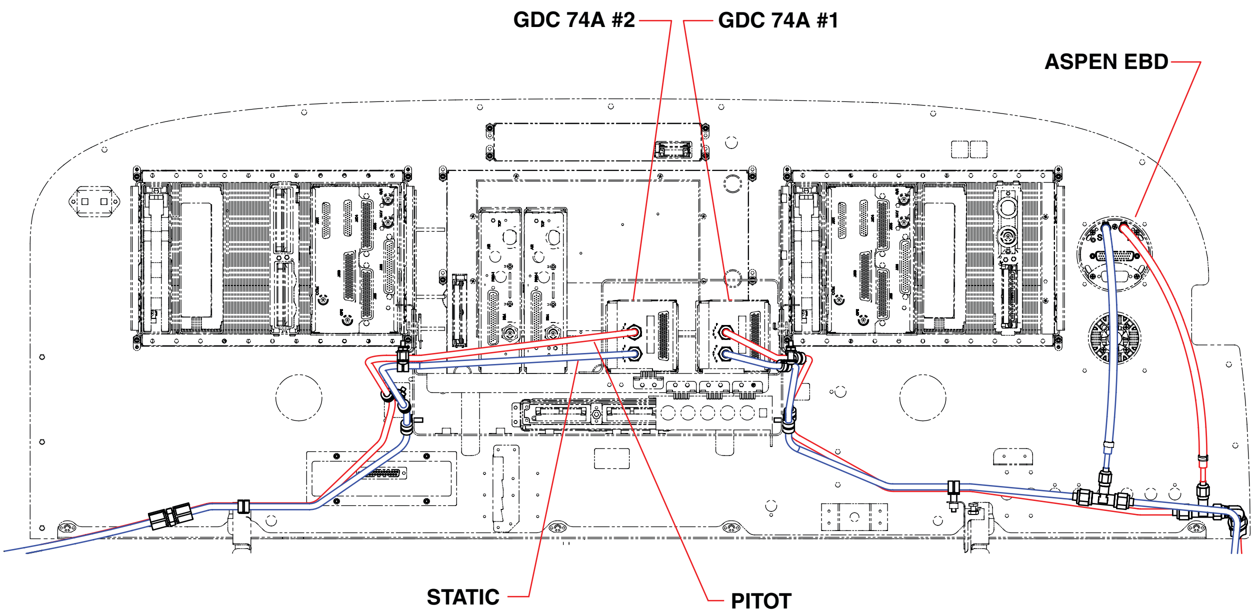
Effectivity
4636633, 4636652–4636685

Pitot-Static System Installation
Figure 1 (Sheet 7 of 12)



Pitot-Static System Installation
Figure 1 (Sheet 8 of 12)

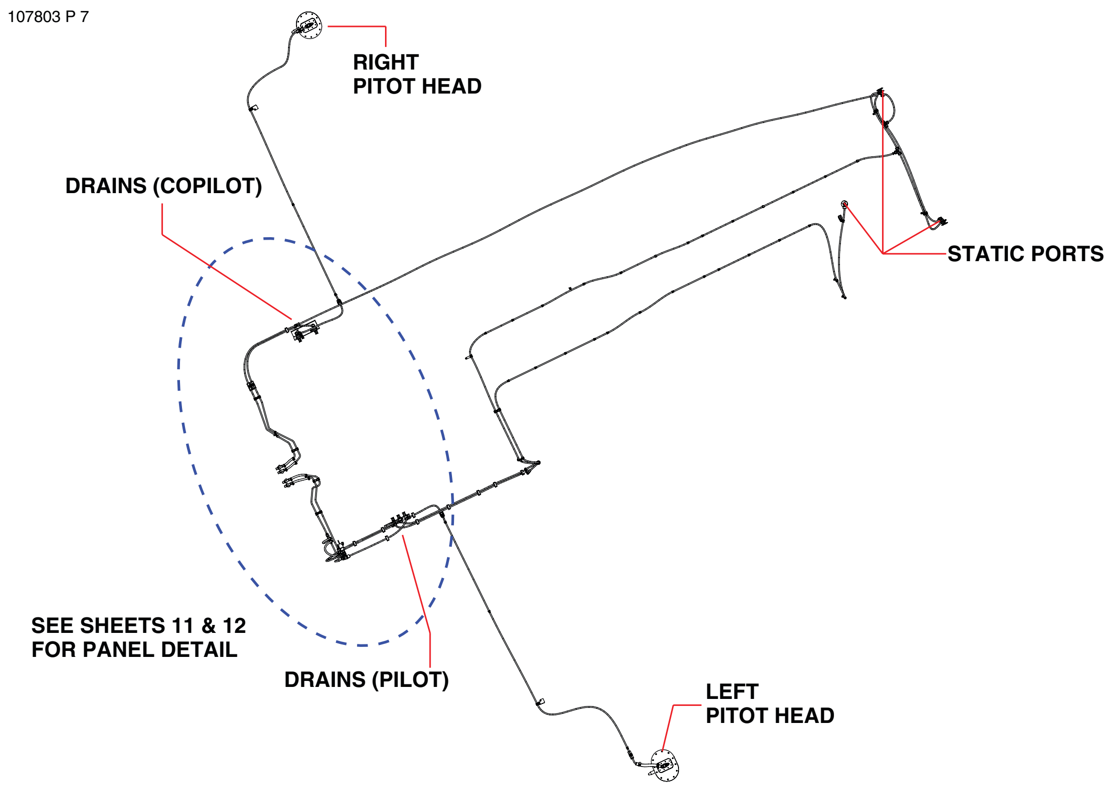
Effectivity
4636686–4636715,
4636717–4636719

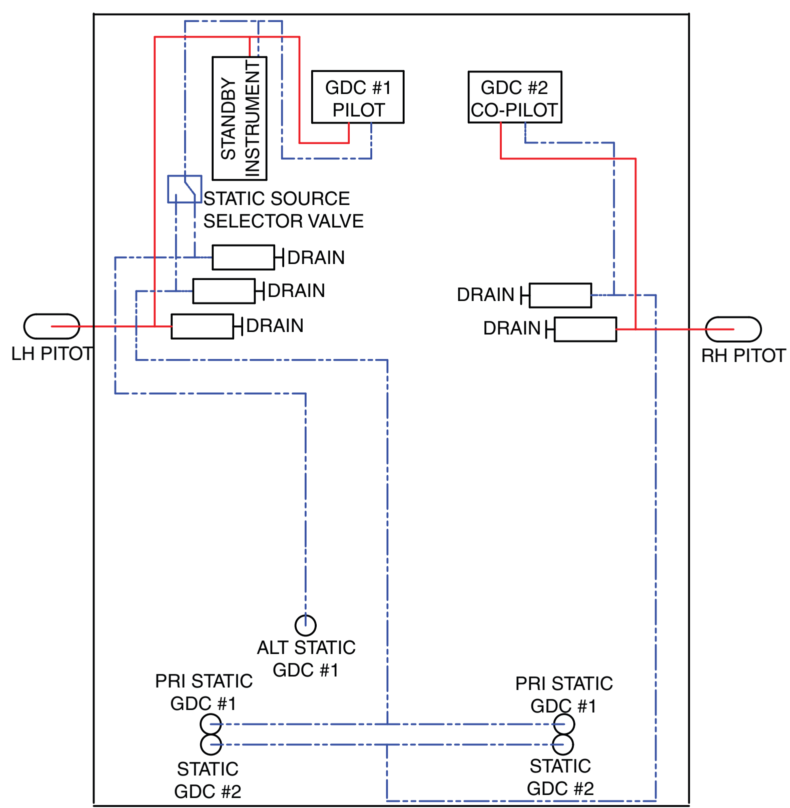


Effectivity
4636633, 4636652–4636715,
4636717–4636719

Pitot-Static System Installation
Figure 1 (Sheet 9 of 12)

341010

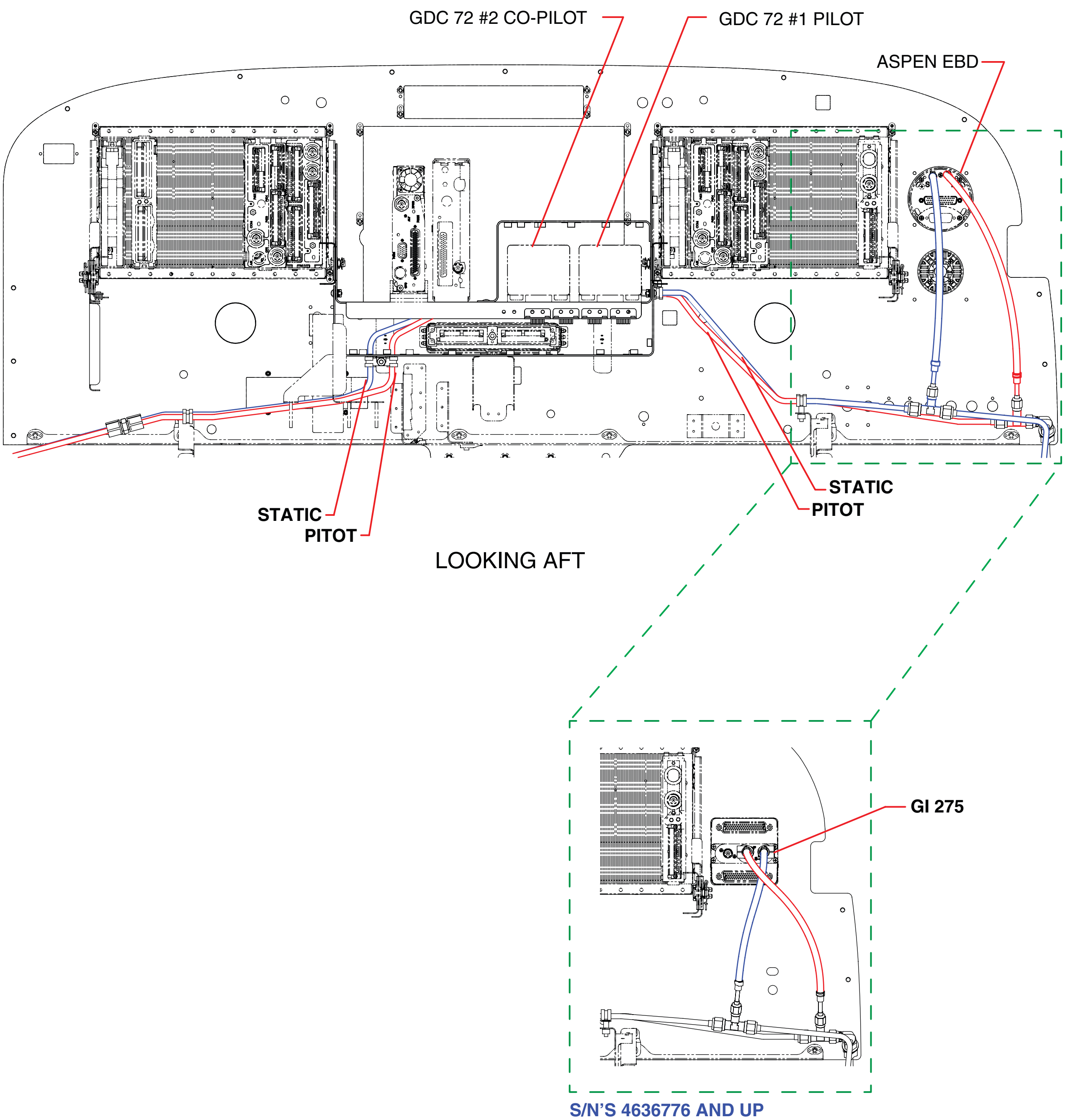




Pitot-Static System Installation
Figure 1 (Sheet 10 of 12)

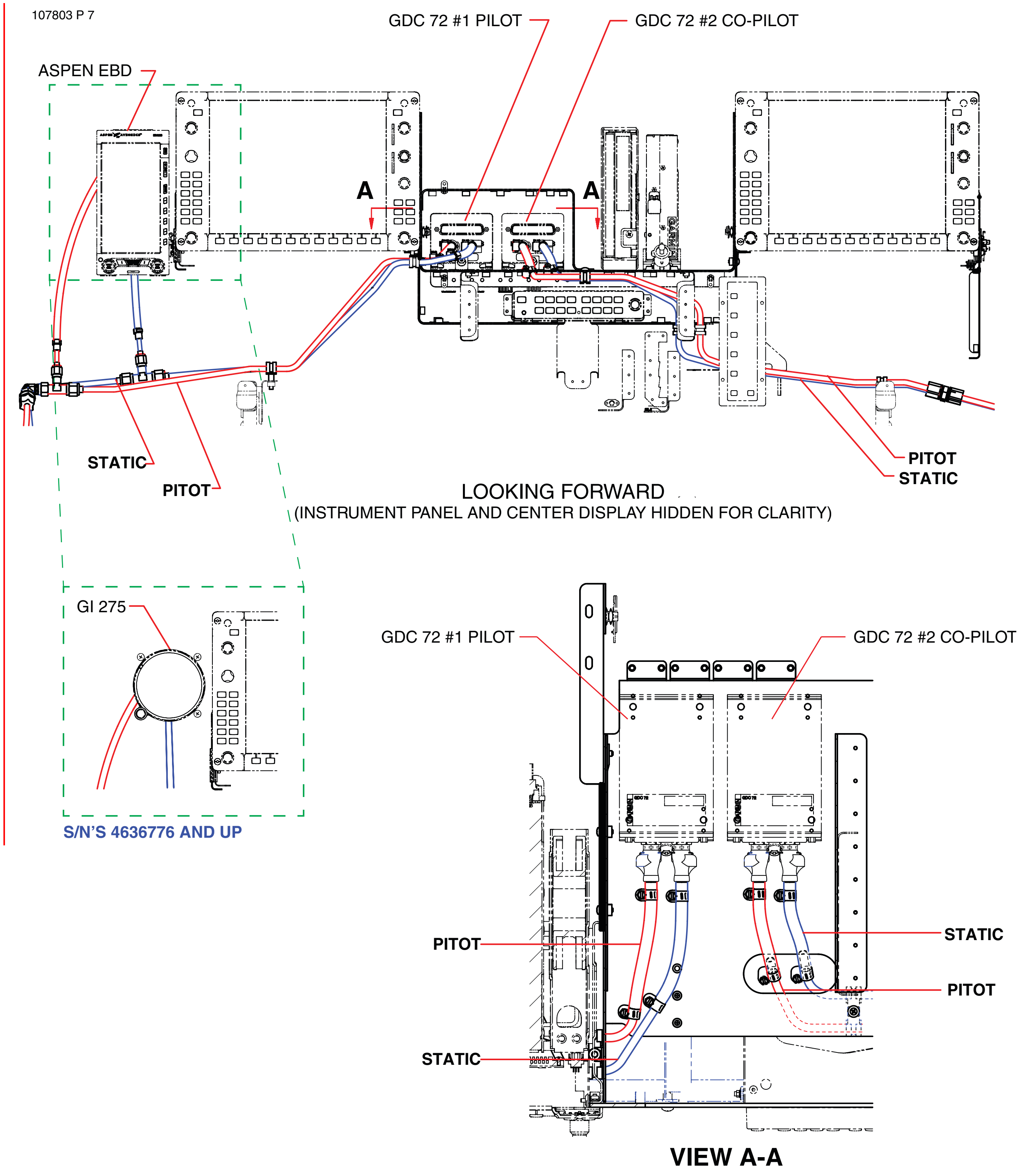
Effectivity
4636716, 4636720 and up

107803 P 7



Effectivity
4636716, 4636720 and up

Pitot-Static System Installation
Figure 1 (Sheet 11 of 12)



Pitot-Static System Installation
Figure 1 (Sheet 12 of 12)

Effectivity
4636716, 4636720 and up

The holes in the sensors for pitot and static pressure must be fully open and free from blockage. Blocked sensor holes will give erratic or zero readings on the instruments.

The heated pitot head(s) is(are) standard equipment. The pitot heat switch is located on the right overhead switch panel. Static source pads have been demonstrated to be non-icing; however, in the event that icing does occur, selecting the alternate static source will alleviate the problem.

B. Troubleshooting

See "Chart 1". With the optional Meggitt EFIS installation, see also, Appendix 1.

C. Test

(PIR-PPS60035, Rev. AD.)

This test requires a pitot/static test fixture (i.e. - Aerosonic Air Data Test Set - Model 90000-0168; Barfield P/N 101-00164, 101-00164A0A or 101-01170; Tactair 381-C; or equivalent) and calibrated air source (i.e. - airspeed simulator) and should be performed at any time an instrument, fitting, line, pitot head, or static button is disconnected. The test should be performed prior to the next flight.

NOTE: If equipped with Garmin G1000 or G1000NXi, see "GDC 74A Air Data Computer Tests" on page 342501188, or "GDC 72 Air Data Computer Tests" on page 342502188, as appropriate, before beginning this test.

NOTE: Ensure the lines and fittings are free of any entrapped moisture or restrictions.

(1) Supply external power (see 24-40-00) to airplanes equipped with Garmin avionics.
(2) Set airplane barometer (i.e., altimeter) to 29.92 In·Hg.
(3) Attach the test fixture to the pitot head. Align the holes in the fixture with the holes in the head.
(4) Attach the airspeed simulator hose to the pitot (pressure) port of the fixture.
(5) When equipped with the optional Meggitt EFIS, turn ON the PFD (i.e., EADI).

CHART 1
TROUBLESHOOTING PITOT AND STATIC SYSTEMS

Trouble	Cause	Remedy
Heating element inoperative.	Defective switch.	Replace the switch.
	Grounded or open circuit.	Check for continuity and repair.
	Defective heating element in pitot head.	Replace the pitot-static mast.
Circuit breaker keeps tripping.	Grounded wire.	Check for continuity and repair.
Instruments inoperative or erratic in operation.	Lines clogged.	Disconnect lines at instruments and blow out with low pressure air.
	Line leaks.	Check lines for loose connections at all connection points.
NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, Test, above. With optional Meggitt EFIS installation, see Appendix 1.		

(6) When equipped with Avidyne Entegra or Garmin G1000/G1000 NXi, turn ON the PFD('s).

(7) Operate the simulator to obtain a reading of 75 knots on the airplane airspeed indicator(s).

(8) Check that the airspeed indicator needles (or PFD airspeed) follow in the same direction as the simulator airspeed indicator needle.

(9) Raise airspeed to 198 knots and wait 15 seconds to allow the airplane airspeed indicators to stabilize.

(10) Observe the simulator and airplane airspeed indicators for 15 seconds. If a leak is present, the indicator needles (or PFD airspeed) will move toward zero. Maximum acceptable leak rate is one (1) knot in 15 seconds.

(11) If a leak rate greater than one (1) knot in 15 seconds is present, check the fixture installation, hose connections, and pitot system lines and fittings. Repair the leak when found, then repeat steps (1)- (10), above.

(12) In S/N's 4636001 thru 4636374 with standard instrumentation, operate the simulator to indicate 140 knots on the airspeed indicators. Verify that the airspeed indicators show within three (3) knots of the simulator indication.

(13) When equipped with the optional Meggitt EFIS:

(a) Operate the simulator to indicate 140 knots on the PFD. Verify that the airspeed indicators show within three (3) knots of the simulator indication on the PFD.

(b) Operate the simulator until the overspeed aural alarm first sounds. Verify that the simulator indicates an airspeed of 198 to 201 knots.

(c) Increase the simulator airspeed indication to 211 knots.

(d) Decrease the simulator airspeed indication until the overspeed aural alarm first stops. Verify that the simulator indicates an airspeed of less than 198 knots.

(e) Turn OFF the PFD (i.e., EADI).

(14) When equipped with Avidyne Entegra:

(a) Turn OFF the copilot's PFD (if installed).

(b) Operate the simulator to indicate 140 knots on the pilot's PFD. Verify that the pilot's PFD indicates within three (3) knots of the simulator indication.

(c) Turn OFF the pilot's PFD and turn ON the copilot's PFD (if installed).

(d) Repeat step (b) for the standby airspeed indicator and for the copilot's PFD (if installed).

(e) Turn ON the pilot's PFD and turn OFF the copilot's PFD (if installed).

(f) Operate the simulator until the overspeed alarm (PFD visual) first activates. Verify that the simulator indicates an airspeed of 198 to 201 knots.

(g) Increase the simulator airspeed indication to 211 knots.

(h) Decrease the simulator airspeed indication until the overspeed alarm (PFD visual) first stops. Verify that the simulator indicates an airspeed of less than 198 knots.

(i) If the copilot's PFD is installed, turn OFF the pilot's PFD and turn ON the copilot's PFD. Repeat steps (f) - (h).

(15) When equipped with Garmin G1000/G1000 NXi:

(a) Turn OFF the copilot's PFD.

(b) Operate the simulator to indicate 140 knots on the pilot's PFD. Verify that the pilot's PFD indicates within three (3) knots of the simulator indication.

(c) Turn OFF the pilot's PFD and turn ON the copilot's PFD.

(d) Repeat step (b) for the standby airspeed indicator and for the copilot's PFD.

(e) Turn ON the pilot's PFD and turn OFF the copilot's PFD.

(f) Operate the simulator until the overspeed alarm (PFD visual/aural) first activates. Verify that the simulator indicates an airspeed of 198 to 201 knots.

(g) Increase the simulator airspeed indication to 211 knots.

(h) Decrease the simulator airspeed indication until the overspeed alarm (PFD visual/aural) first stops. Verify that the simulator indicates an airspeed of less than 198 knots.

(i) Pull the PFD1 circuit breaker.

(j) Turn OFF the pilot's PFD and turn ON the copilot's PFD. Repeat steps (f) - (h).

(k) Reset all PFD circuit breakers.

(16) Remove the test fixture from the pitot head.

(17) Attach static test fixture to the static port and pitot test port of the aircraft. Tape over the other static button.

(18) When equipped with the optional Meggitt EFIS, turn ON the PFD (i.e. - EADI).

(19) When equipped with Avidyne Entegra or Garmin G1000/G1000 NXi, turn ON the Pilot's PFD.

(20) Set the aircraft altimeter needles to read zero altitude. Operate the static simulator to cause the aircraft altimeter needles (or PFD airspeed) to read 1,000 feet altitude. Momentarily open the alternate static port. There should be a decrease in altimeter indication. If no change occurs, the system is blocked and must be repaired prior to further testing.

(21) In PA-46R-350T, increase altitude to 1,050 feet. In PA-46-350P, increase altitude to 1,000 feet.

NOTE: In PA-46-350P, increase the altitude to 15,000 feet at a minimum climb rate of 3,000 ft. per minute before proceeding to the leak check at 12,700 feet. This will allow the max. pressure differential valve inside the outflow valve to actuate, releasing the pressure in the outflow valve's reference chamber and allowing the pressure in the line between the outflow valve and the static port to equalize.

(22) Check that the aircraft altimeter shows an increase.

(23) Observe the aircraft altimeter. Loss of indicated altitude shall not exceed 100 feet in one minute.

NOTE: For pressurized PA-46 models perform preliminary leak check at 1000 feet before proceeding to 12,700 feet.

(24) For pressurized aircraft, increase altitude to 12,700 feet.

(25) Check that the aircraft altimeter shows an increase.

(26) Observe the aircraft altimeter. For pressurized PA-46 models loss of altitude shall not exceed 100 feet in one minute at 1000 feet, or 254 feet in one minute at 12,700 feet.

(27) If a leak exceeds the tolerances in step (23) or (26), check the fixture installation, plumbing and fittings. Repair the leak when found and repeat the static system checks above.

(28) When equipped with the optional Meggitt EFIS, turn OFF the PFD (i.e. - EADI).

(29) When equipped with dual PFD Avidyne Entegra or Garmin G1000/G1000 NXi, turn OFF the Pilot's PFD and turn ON the copilot's PFD. Repeat steps (18) thru (25) for copilot's PFD.

(30) Remove the test fixture and tape from the static button.

(31) In S/N's 4636633, 4636652 and up only:

(a) Attach a static test device to one of the the cabin pressure control system's (CPCS) stand-alone static ports. See "Figure 2".

(b) Increase the altitude to 1,000 feet. Leak rate shall be less than 100 ft/min.

(c) Increase the altitude to 15,000 feet at a minimum climb rate of 3,000 ft/min. This will allow the max pressure differential valve inside the outflow valve to actuate, release the pressure in the outflow valve's reference chamber, allowing the pressure in the line between the outflow valve and the static port to equalize. Decrease the altitude to 12,700 feet. Leak rate shall be less that 254 ft/min.

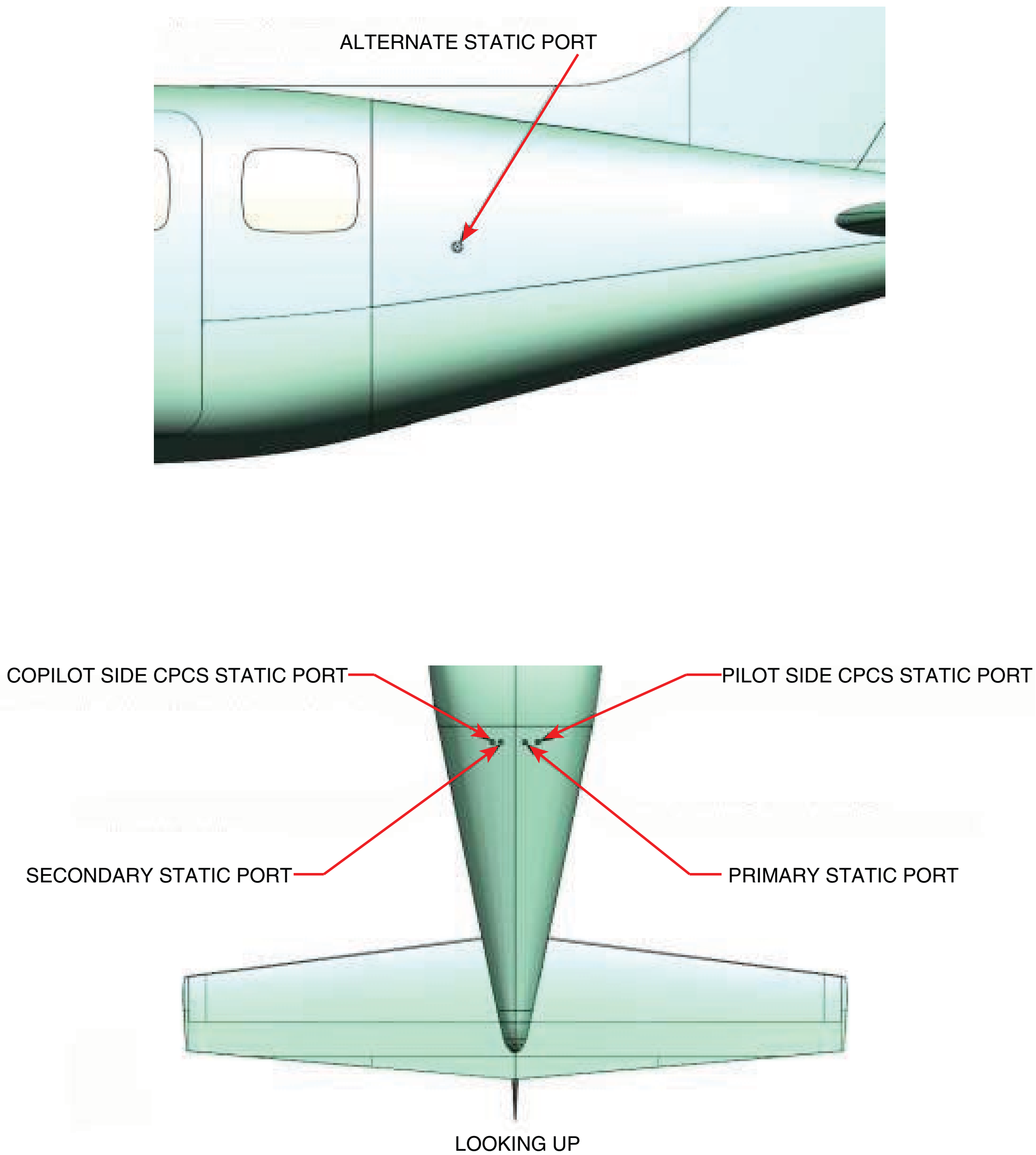
(d) If a leak exceeds the above tolerance, check fixture installation, plumbing and fittings. Repair leak when found, and repeat static system checks above.

NOTE: If a leak exceeds the tolerance at 12,700 feet but passes at 1,000 feet, check for leaks in the cabin pressure control system lines between the cabin pressure controller and the outflow valves.

(e) Repeat steps (b) thru (d) for other CPCS static port.

(f) Remove the static test device from the CPCS static port if applicable.

(32) Remove external power from aircraft if previously applied.



CPCS and Static Port Locations
Figure 2

2. Pitot Head

See "Figure 3" on page 341019.

A. S/N's 4636001 thru 4636374 and S/N's 4692001 and up

The pitot head is located on the underside of the left wing.

(1) Removal

(a) Remove the four screws which secure the pitot tube to the mast assembly.

(b) Carefully pull the pitot tube from the mast assembly.

(c) Remove the hose from the elbow on top of the pitot tube.

(d) Disconnect the electrical leads at the connector.

(2) Installation

(a) Reconnect the electrical leads to the heating elements.

(b) Install the pitot tube into the mast assembly.

(c) Secure the pitot tube to the mast assembly with the four screws.

(d) Reconnect the electrical leads to the gauge.

(3) Heaters Test

(a) Remove the pitot head from mast assembly and the wing.

(b) With one of the test leads connected to one of the terminals of the head, ground the other lead to the metal body of the head assembly. There should be no continuity.

(c) Reinstall the pitot head into the mast assembly and the wing.

B. S/N's 4636375 and up

The pitot head is located on the underside of each wing installed on a access panel.

(1) Removal

(a) Remove the twelve (12) screws that secure the pitot assembly to the wing. Support the assembly as it comes free from the wing.

(b) Carefully disconnect the pitot tube and the electrical connector from the pitot assembly.

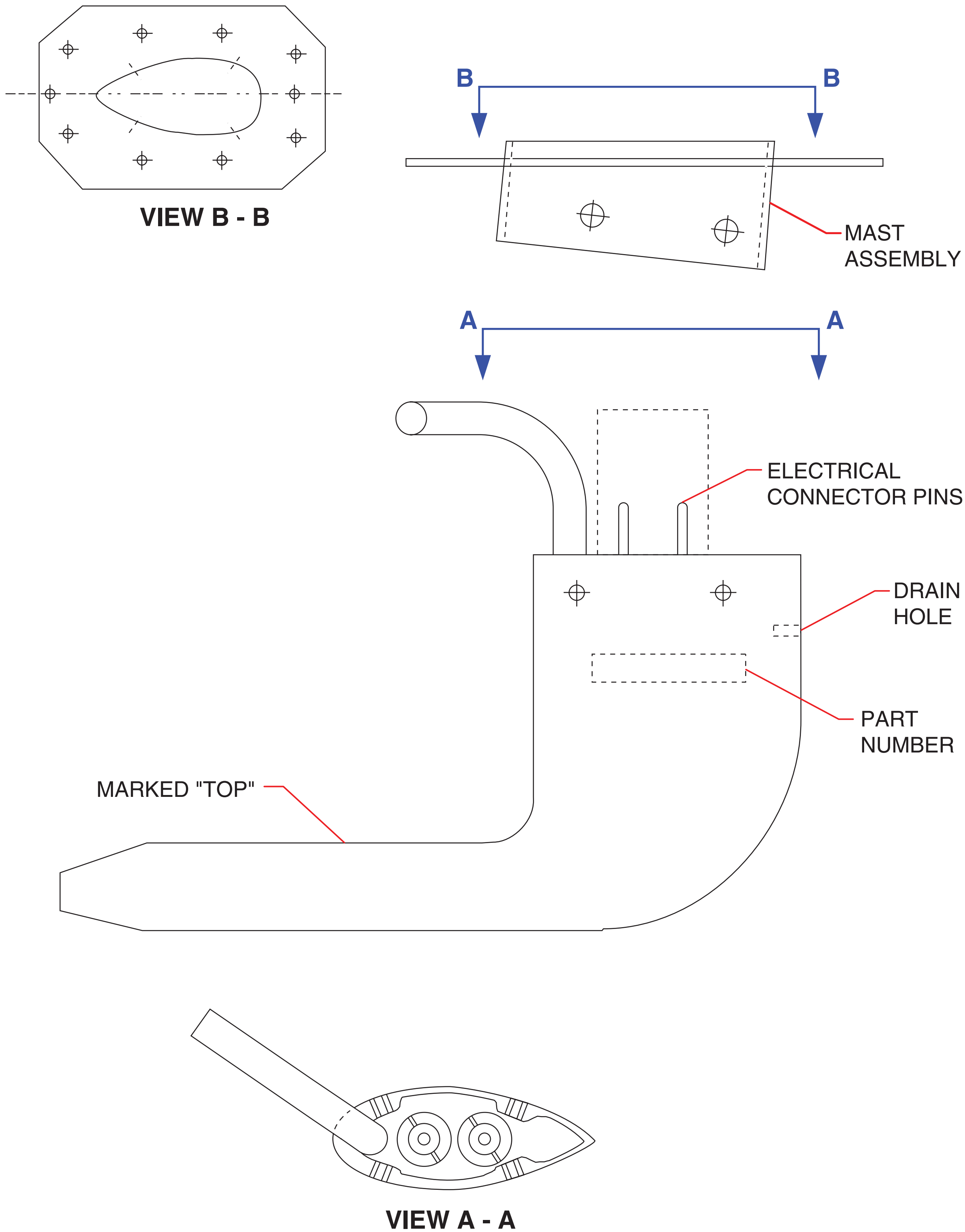
(c) If required, remove the four (4) screws, washers, bolts and the doubler to separate the pitot probe from the access panel. Clean off fillet seal material from the probe and panel.

(2) Installation

(a) If the pitot probe and access panel were separated during removal, secure the probe to the access panel using the four (4) screws, washers, bolts, and doubler. Fillet seal around the periphery of the pitot head and access panel using an approved manganese oxide cured polysulfides sealant per SAE AMS-S-8802 Type 2 Classes A and B.

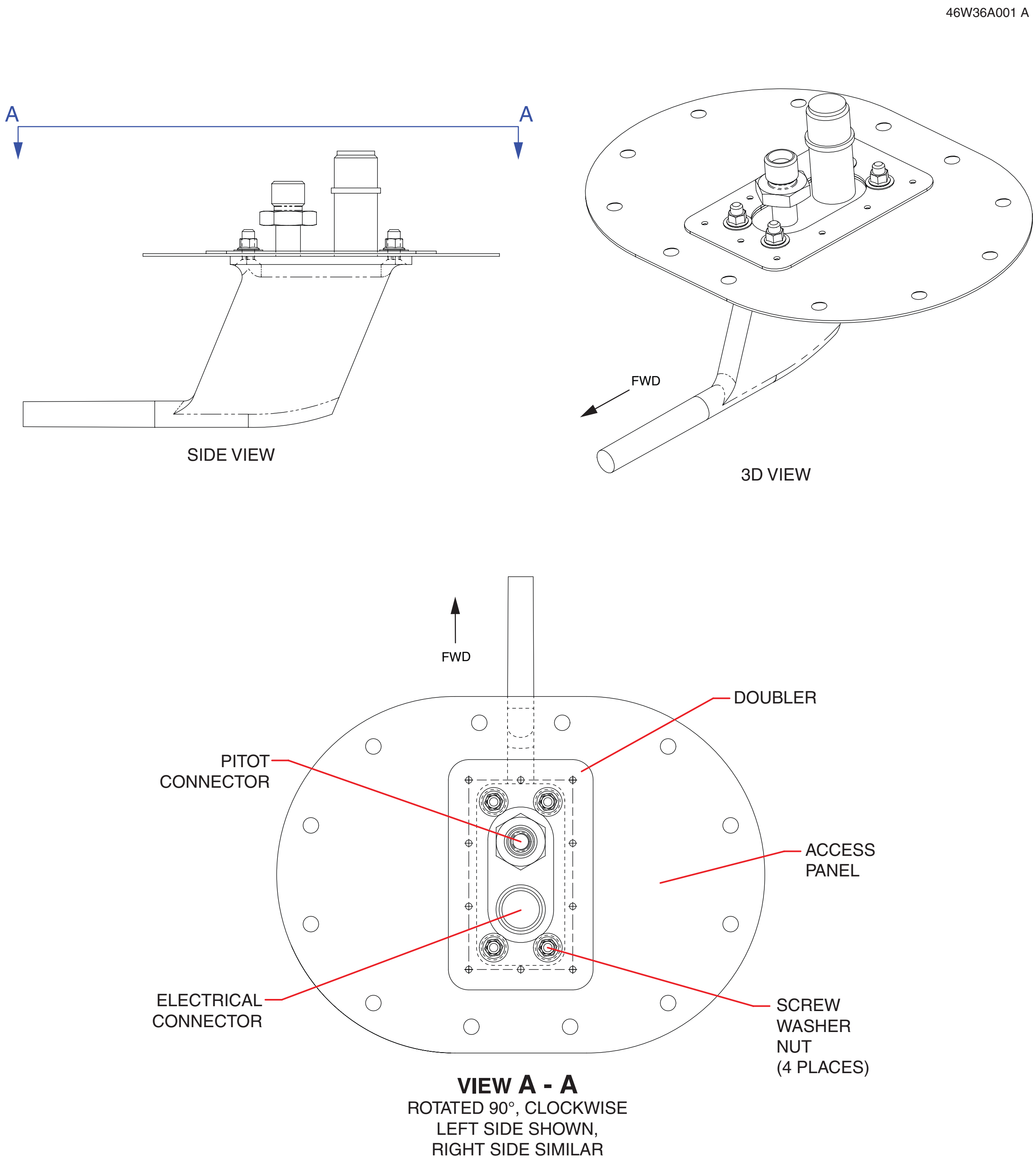
(b) Connect the pitot tube and electrical connector to the pitot probe.

(c) Secure the pitot assembly to the wing with the twelve (12) screws. Support the assembly while securing the assembly.



Pitot Tube Installation
Figure 3 (Sheet 1 of 2)

Effectivity
4636001–4636759



Effectivity
4636760 and up

"L" Shaped Pitot Tube Installation
Figure 3 (Sheet 2 of 2)

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3. Vertical Speed Indicator

NOTE: In airplanes equipped with either Avidyne Entegra or Garmin G1000/G1000 NXi, this function is performed solely by the Primary Flight Display (PFD).

The Vertical Speed indicator measures the rate of change in static pressure when the airplane is climbing or descending. By means of a pointer and dial this instrument will indicate a rate of ascent or descent of the airplane in feet per minute. But due to the lag of the instrument, the aircraft will be climbing or descending before the instrument starts to read and the instrument will continue to read after the aircraft has assumed level flight. This lag should not be considered a malfunction when operating in turbulent (rough) air.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 341014. With optional Meggitt EFIS installation, see Appendix 1.

A. Troubleshooting

See "Chart 2".

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

4. Altimeter / Standby Altimeter

NOTE: In airplanes equipped with either Avidyne Entegra or Garmin G1000/G1000 NXi, the primary altitude indicator is the Primary Flight Display (PFD). A conventional Standby Altimeter was also provided into 2014.

With all Avidyne Entegra and PA-46R-350T Garmin G1000 installations the standby altimeter is a standalone unit described here. In PA-46-350P Garmin G1000/G1000 NXi installations, early (2010–2014) airplanes are equipped with the standalone unit described here. In late 2014–2020, the standby altimeter is part of the Aspen Avionics EBD, see page 34241. In 2021 and later, a Garmin GI 275 is installed, see page 3424011.

The altimeter uses pressure differential to indicate altitude in feet. When the barometric pressure scale on the instrument is set to the current altimeter setting, the altimeter will indicate altitude, in feet, above sea level. The indicator has three pointers and a dial scale; the long pointer is read in hundreds of feet, the middle pointer in thousands of feet and the short pointer in tens of thousands of feet. A window in the lower center of the dial face contains a “barber pole” design to show when the airplane is below 10,000 feet. As the airplane climbs, the barber pole design progressively disappears until, when at or above 10,000 feet, the window is blank. The barometric pressure window, located on the right side of the indicator dial, is set by the knob located on the lower left corner of the instrument. The altimeter consists of a sealed aneroid (diaphragm) that is connected to the pointers through a mechanical linkage. The instrument case is vented to the static air system and as static air pressure decreases, the aneroid expands, causing the pointers to move through the mechanical linkage.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 341014. With optional Meggitt EFIS installation, see Appendix 1.

A. Troubleshooting

See "Chart 3" on page 341024.

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 2
TROUBLESHOOTING VERTICAL SPEED INDICATOR

Trouble	Cause	Remedy
Pointer does not set on zero.	Aging of diaphragm.	Reset pointer to zero by means of setting screw. Tap instrument while resetting.
Pointer fails to respond.	Obstruction in static line.	Disconnect all instruments connected to the static line.
		Clear line.
	Pitot head frozen over.	Apply pitot heat.
	Water in static line.	Check individual instruments for obstruction in lines.
	Obstruction in pitot head.	Clean lines and head.
Pointer oscillates when not operating in turbulent air.	Leak in static lines.	Disconnect all instruments connected to the static line.
		Check individual instruments for leaks. Reconnect instruments to static line and test installation for leaks.
	Defective mechanism.	Replace instrument.
Vertical Speed indicates when aircraft is banked.	Water in static line.	Disconnect static lines and blow out lines from cockpit out to static ports.
Pointer has to be set before every flight.	Temperature compensator inoperative.	Replace instrument.
Pointer cannot be reset to zero.	Diaphragm distorted.	Replace instrument.
Instrument reads very low during climb or descent.	Case of instrument broken or leaking.	Replace instrument.

CHART 3
TROUBLESHOOTING ALTIMETER

Trouble	Cause	Remedy
Excessive scale error.	Improper calibration adjustment.	Replace instrument.
Excessive pointer oscillation.	Defective mechanism.	Replace instrument.
High or low reading.	Improper venting.	Eliminate leak in static pressure system and check alignment of airspeed tube.
Setting knob is hard to turn.	Wrong lubrication or lack of lubrication.	Replace instrument.
Inner reference marker fails to move when setting knob is rotated.	Out of engagement.	Replace instrument.
Setting knob set screw loose or missing.	Not tight when altimeter was reset.	Tighten instrument screw, if loose. Replace instrument, if screw is missing.
Cracked or loose cover glass.	Case gasket hardened.	Replace instrument.
Dull or discolored markings.	Age.	
Barometric scale and reference markers out of synchronism.	Slippage of mating parts.	Replace instrument.
Barometric scale and reference markers out of synchronism with pointers.	Drift in mechanism.	Refer to the latest revision of AC 43.13-1.
Altimeter sticks at altitude or does not change with change of altitude.	Water or restriction in static line.	Remove static lines from all instruments, blow line clear from cockpit to static ports.
Altimeter changes reading as aircraft is banked.	Water in static line.	Remove static lines from all instruments, and blow line clear from cockpit to static ports.
Altimeter requires resetting frequently.	Temperature compensator inoperative.	Change instrument.

5. Airspeed Indicator / Standby Airspeed Indicator

NOTE: In airplanes equipped with either Avidyne Entegra EFIS or Garmin G1000/G1000 NXi IAS, the primary airspeed indicator is the Primary Flight Display (PFD). A conventional Standby Airspeed Indicator was also provided into 2014.

With all Avidyne Entegra and PA-46R-350T Garmin G1000 installations the standby airspeed indicator is a standalone unit described here. In PA-46-350P Garmin G1000/G1000 NXi installations, early (2010–2014) airplanes are equipped with the standalone unit described here. In 2014–2020, the standby airspeed indicator is part of the Aspen Avionics EBD, see page 34241. In 2021 and later, a Garmin GI 275 is installed, see page 3424011.

The airspeed indicator provides a means of indicating the speed of the airplane passing through the air. The airspeed indication is the differential pressure reading between pitot air to pressure and static air pressure. This instrument has the diaphragm vented to the pitot air source, and the case is vented to the static air system. As the airplane increases speed, the pitot air pressure increases, causing the diaphragm to expand. A mechanical linkage picks up this motion and moves the instrument pointer to the indicated speed. The instrument dial is calibrated in knots, and also has the necessary operating range markings for safe operation of the airplane.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 341014. With optional Meggitt EFIS installation, see Appendix 1.

A. Troubleshooting

See "Chart 4".

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 4
TROUBLESHOOTING AIRSPEED TUBES AND INDICATOR

Trouble	Cause	Remedy
Pointers on instruments stick or do not indicate properly.	Leak in instrument case or in pitot lines.	Check for leak and seal.
Pointer of instrument oscillates.	Defective mechanism.	Replace instrument.
Instrument reads high.	Pointer not on zero.	Replace instrument.
	Leaking static system.	Find leak and correct.
Instrument reads low.	Pointer not on zero.	Replace instrument.
	Leaking static system.	Find leak and correct.
	Pitot head not aligned correctly.	Realign pitot head.
Airspeed changes as aircraft is banked.	Water in pitot line.	Remove lines from static instruments and blow out lines from cockpit to pitot head.

6. Outside Air Temperature (OAT)

A. S/N's 4636001 thru 4636020

The OAT indicator is located just forward of the pilot's side window at F.S. 122.30 (see "Figure 4"). The probe of the direct reading gauge screws into a bracket and protrudes through the airplane's skin.

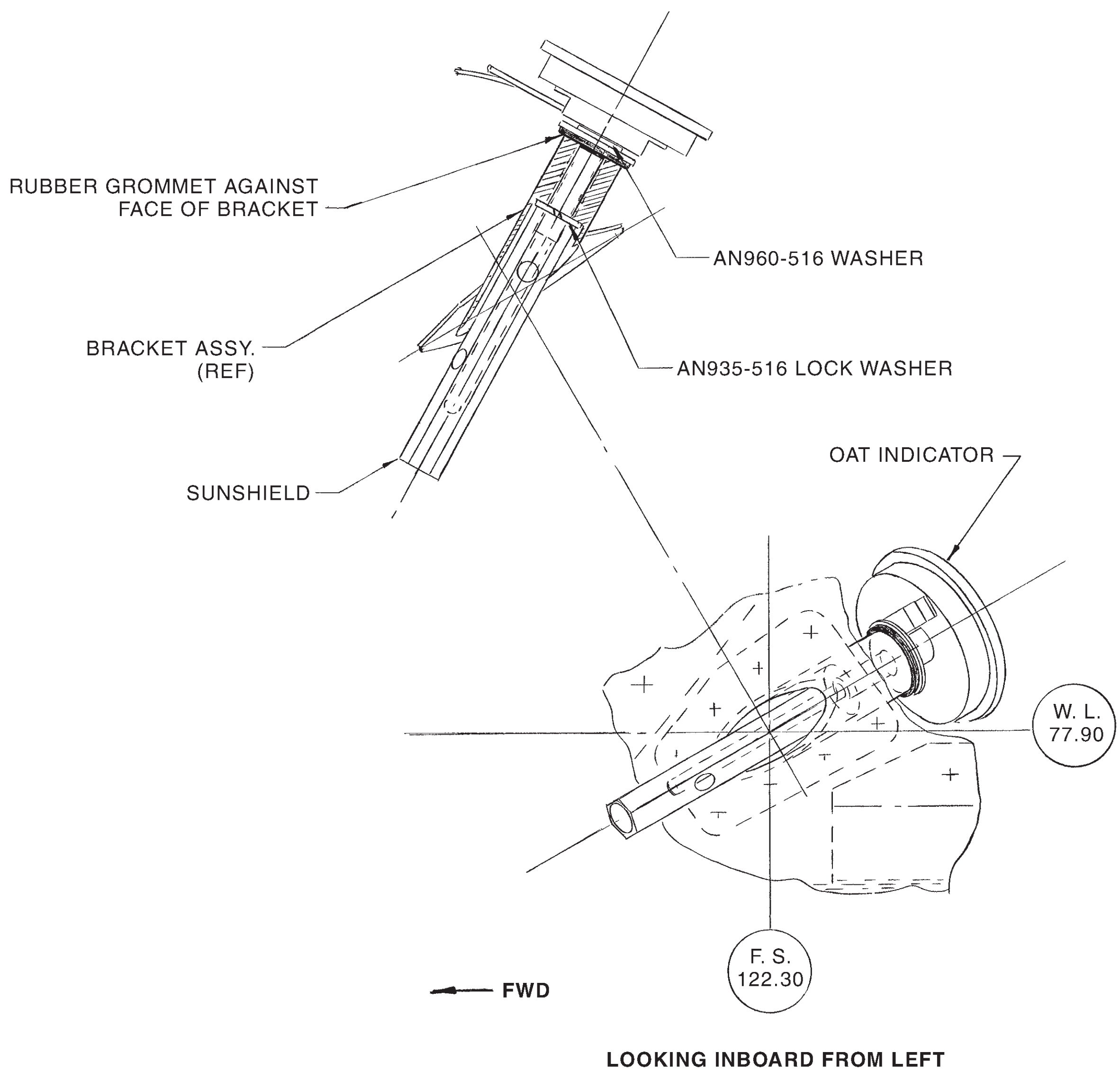
(1) Removal

(a) Disconnect the electrical leads from the gauge.

(b) Turn the gauge counterclockwise to screw it out of the sunshield. It may be necessary to have an assistant hold the sunshield to prevent it from turning with the gauge.

(c) Remove the gauge, washers, and rubber grommet from inside the airplane.

(d) Remove the sunshield and lock washer from outside the airplane.



Effectivity
4636001 thru 4636020

Outside Air Temperature Indicator
Figure 4

(2) Installation

(a) Place the two washers and rubber grommet over the stem of the gauge.

(b) Slide the gauge into its support bracket in the airplane.

(c) Slide the lock washer and sunshield over the stem of the gauge (from the outside).

(d) Screw the gauge clockwise into the sunshield.

(e) Reconnect the electrical leads to the gauge.

B. S/N's 4636021 thru 4636374

A remote OAT probe is located on the underside of the right wing (see "Figure 5"):

- mounted to a cover plate at W.S. 86.50 in S/N's 4636021 thru 4636369; and,
- mounted in the wing skin at W.S. 244.3 in S/N's 4636370 thru 4636374.

OAT measurement is accomplished by an OAT sensor and the Transicoil Electronic Module Instrument System (EMIS). See 77-40-00 for more information.

C. S/N's 4636375–4636459, 4636461–4636462, 4636481; and 4692001–4692133, 4692141, 4692149, 4692153 (i.e., with Avidyne Entegra)

A remote OAT probe is located on the underside of the right wing (see "Figure 5") at W.S. 244.3.

The Data Acquisition Unit (DAU) collects the OAT sensor data and sends it to the Primary Flight Displays (PFDs) and the Multifunction Display (MFD).

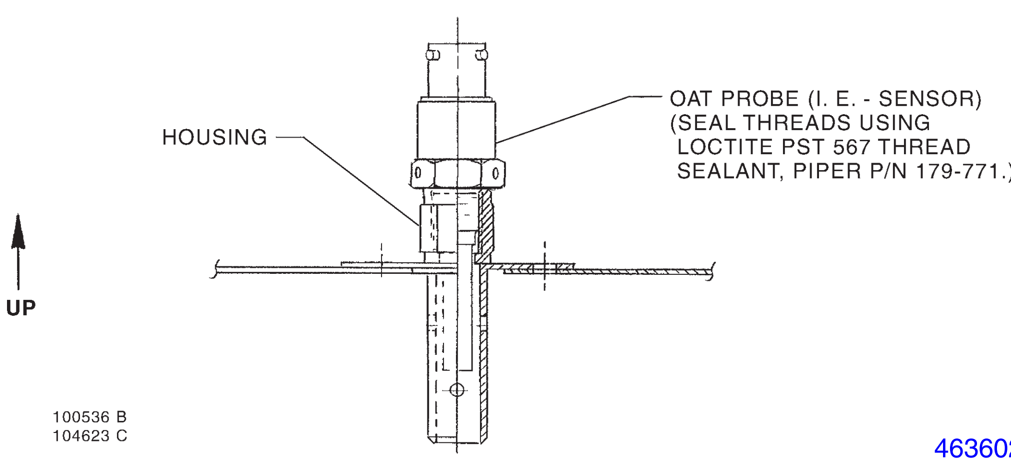
See Electronic Flight Instrument System - Avidyne in 34-20-00 for additional information.

D. S/N's 4636460, 4636463 and up less 4636481; and 4692134 and up, less 4692141, 4692149, 4692153 (i.e., with Garmin G1000/G1000 NXi).

The dual GTP 59 OAT probes are located on the underside of the left wing (see "Figure 6" on page 341028).

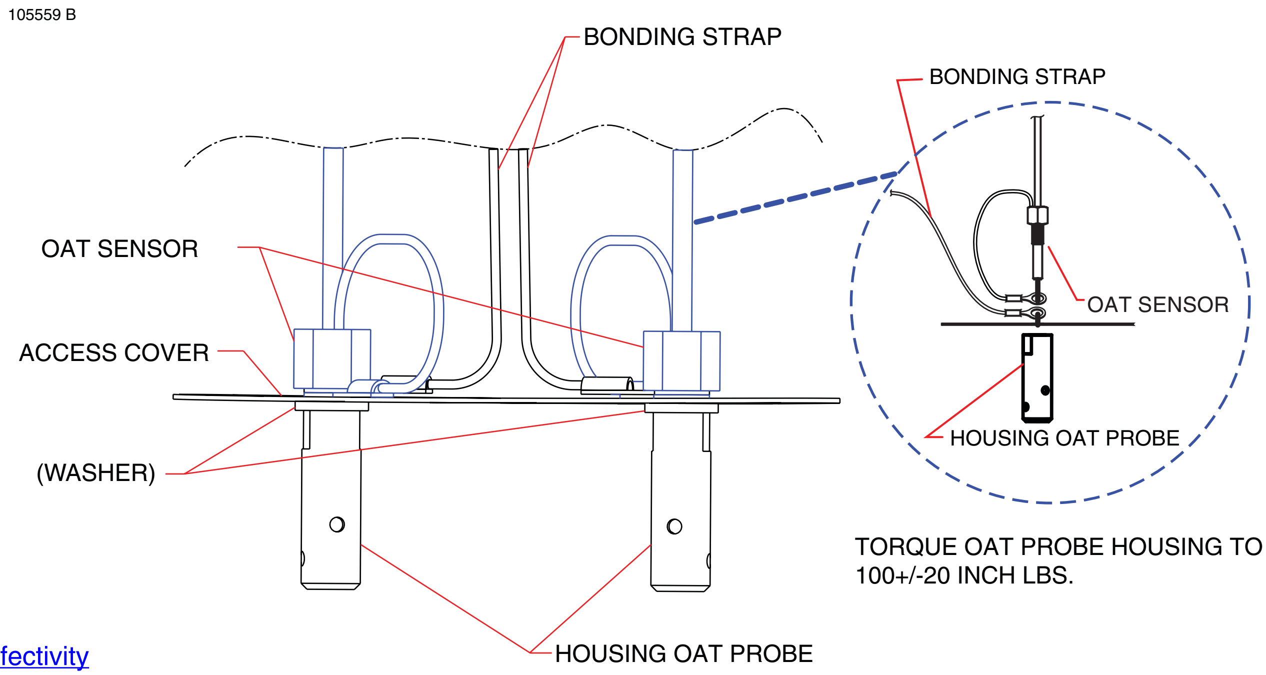
The OAT probes supply data to the GDC 74A/GDC 72 air data computer that provides information to the G1000/G1000 NXi system.

See Integrated Avionics System - Garmin, in 34-25-01 (G1000) or 34-25-02 (G1000 NXi), for additional information.



Effectivity
4636021–4636459
4636461–4636462, 4636481
4692001–4692133, 4692141
4692149 and 4692153

OAT Probe Installation
Figure 5



Effectivity
4636460, 4636463 and up
less 4636481
4692134 and up, less 4692141
4692149 and 4692153

OAT Probe Installation
Figure 6

ATTITUDE AND DIRECTION

1. Attitude Indicator

NOTE: In airplanes equipped with either Avidyne Entegra or Garmin G1000/G1000 NXi, no air-driven attitude indicator is installed. In those installations, this function is provided by the Primary Flight Display (PFD).

The Attitude Indicator is essentially an air-driven gyroscope rotating in a horizontal plane and is operated on the principal of rigidity in space. Due to the gyroscopic inertia, the spin axis continues to point in the near vertical direction, providing a constant visual reference to the attitude of the airplane relative to pitch and roll axes. A bar across the face of the indicator represents the horizon, and aligning the miniature airplane to the horizon bar simulates the alignment of the airplane to the actual horizon. Any deviation simulates the deviation of the airplane from the true horizon. The Attitude Indicator is marked for different degrees of bank.

A. Troubleshooting

See "Chart 1".

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 1
TROUBLESHOOTING ATTITUDE INDICATOR

Trouble	Cause	Remedy
Bar fails to respond.	Insufficient vacuum.	Check pump and tubing.
	Vacuum System Filter dirty.	Clean or replace filter.
Bar does not settle.	Insufficient vacuum.	Check line and pump. Adjust valve.
	Incorrect instrument.	Check part number.
	Defective instrument.	Replace.
Bar oscillates or shimmies continuously.	Instrument loose in panel.	Tighten mounting screws.
	Vacuum too high.	Adjust valve.
	Defective mechanism.	Replace instrument.
Instrument does not indicate level flight.	Instrument not level in panel.	Loosen screws and level instrument.
	Aircraft out of trim.	Trim aircraft.
Bar high after 180° turn.	Normal, if it does not exceed 1/16th inch.	Self correcting.
Instrument tumbles in flight.	Low vacuum.	Reset regulator.
	Dirty Vacuum System filter.	Clean or replace filter.
	Line to filter restricted.	Replace line.
	Plug missing or loose in instrument.	Replace or tighten plug.

2. Standby Attitude Indicator

NOTE: In airplanes equipped with either Avidyne Entegra or Garmin G1000/G1000 NXi, the primary attitude indicator is the Primary Flight Display (PFD).

In S/N's 4636375–4636651, less 4636633, and S/N's 4692001 and up with Avidyne Entegra or Garmin G1000 installed, an electric standby attitude indicator is installed to the left of the pilot's Primary Flight Display (PFD).

In S/N's 4636633, 4636652–4636775, the standby attitude indicator is part of the the Aspen Avionics EBD, see page 34241. In S/N's 4636776 and up, a Garmin GI 275 is installed, see page 3424011.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: DO NOT SWITCH THE STANDBY ATTITUDE INDICATOR ON OR OFF WHILE THE AIRPLANE IS MOVING.

CAUTION: WAIT APPROXIMATELY THREE (3) MINUTES AFTER SWITCHING THE STANDBY ATTITUDE INDICATOR ON, BEFORE CAGING THE GYRO TO ENSURE THE GYRO IS UP TO FULL SPEED AND LEVEL.

CAUTION: DO NOT LET THE "PULL TO CAGE" KNOB SNAP BACK INTO POSITION WHEN CAGING THE GYRO. CAGE THE INSTRUMENT BY PULLING THE "PULL TO CAGE" KNOB OUT AND HOLDING IT WHILE OBSERVING THE INDICATOR. WHEN THE INDICATOR APPEARS "STABILIZED," QUICKLY ALLOW THE "PULL TO CAGE" KNOB TO GENTLY RETURN TO ITS NORMAL POSITION.

CAUTION: DO NOT MOVE GYRO UNIT OR THE AIRPLANE WHILE THE GYRO IS SPINNING UP OR DOWN. WHEN INSTALLED IN THE AIRPLANE, DO NOT MOVE THE AIRPLANE FOR A MINIMUM OF FIVE (5) MINUTES AFTER SWITCHING THE STANDBY ATTITUDE INDICATOR OFF. MOVING THE UNIT OR AIRPLANE PREMATURELY MAY DAMAGE THE BEARINGS.

CAUTION: BE EXTREMELY CAREFUL WHEN HANDLING THE UNIT DURING REMOVAL, INSTALLATION, OR AT ANY TIME THE UNIT IS NOT STABILIZED AT FULL OPERATING SPEED. DROPPING A UNIT THAT IS NOT STABILIZED AT FULL OPERATING SPEED FROM A HEIGHT OF AS LITTLE AS ONE (1) INCH CAN DAMAGE THE BEARINGS. LIKEWISE ANY ABRUPT MOVEMENT WHILE HANDLING CAN DAMAGE THE BEARINGS.

Other than removing and replacing the unit itself (see 39-10-00), the only line-replaceable part is the emergency power battery which is located in the aft section of the radar pod (PA-46-350P) or in the rear of the instrument's case (PA-46R-350T).

A. Emergency Power Supply (S/N's 4636375–4636459, 4636461–4636462, 4636481)

CAUTION: MAINTAIN THE EMERGENCY POWER SUPPLY BATTERY AT A MINIMUM 50% CHARGER OR BETTER.

CAUTION: BATTERIES IN STORAGE MUST BE TRICKLE CHARGED.

CAUTION: DO NOT ENGAGE THE "TEST" BUTTON FOR MORE THAN FIVE (5) SECONDS. HOLDING THE "TEST" BUTTON LONGER WILL DEPLETE THE BATTERY.

DO NOT REPEAT THE "TEST" UNTIL THE BATTERY HAS BEEN RECHARGED.

NOTE: Battery will charge with aircraft power "ON."

A 24Vdc battery-operated power supply (i.e. - B.F. Goodrich PS-834A) is installed in the aft radar pod compartment to provide emergency power for the standby attitude indicator. The unit is connected to the main electrical bus and its sealed lead-acid cells are automatically charged and recharged by its built-in charging system.

(1) System Test.

The power supply may be tested locally using its integral test switch and light, or, remotely, using the annunciator (i.e. - STBY ATT IND TEST) and instrument panel mounted remote switch (i.e. - Standby Gyro Test Switch).`

In either case, place the switch in TEST position for five (5) seconds. The test light must remain illuminated the entire time. The built-in-test function checks for operation of the up-converter with an internal load applied. During the built-in-test, the input power is automatically switched off and then the voltage of the battery pack is compared to a temperature-compensated reference. If the battery capacity is greater than 50% and the built-in test monitors are valid, a green “TEST” LED on the connector end of the unit and the remote test lamp will remain on as long as the unit is in test mode.

While in the remote activated built-in-test mode, the 28 Vdc output will maintain 0.24 ± 0.1 Amp at 24 ± 0.2 Vdc for 4.0 ± 1.0 seconds. Six seconds after remote activation, the 28 Vdc output will time-out and drop to 24 - 0.1, + 0.5 Vdc and less than 0.1 Amp.

If the test light fails to illuminate as required during the test, one or more of the following conditions may exist:

(a) The power supply batteries are less than 50% charged;

(b) The test lamp is burned out; or,

(c) The power supply unit is defective.

To determine which condition(s) exist(s), first check the test lamp and then proceed to the Capacity Test, below.

(2) Capacity Test.

Each twelve months, verify the B.F. Goodrich PS-834A Emergency Power Supply battery capacity as follows:

(a) Required Equipment

Power Supply - variable, 0-32 Vdc @ 6.0 amps continuous output.

Voltmeter - Fluke 8050A or equivalent

Load Resistors - 100.0 ohm ± 2%, 10.0 W (28.0 V load)

- 13.0 ohm (or equivalent) ± 2%, 5.0 W (5.0 V load)

(b) Procedure

1) Disconnect and remove power supply from radar pod.

2) Ground J1 pin E or F.

NOTE: Connect directly to the pin(s) or via a test harness equipped with a MS3106F20-27S (or equivalent) connector.

3) Apply 28.0 ± 0.5 Vdc to J1 pin A for two (2) hours.

4) After two (2) hours, remove 28 Vdc input.

5) Apply 100.0 ohm ± 2%, 10.0 W, load resistor to J1 pin C.

6) Apply 13.0 ohm ± 2%, 5.0 W, load resistor to J1 pin H.

NOTE: The 13 ohm resistor can consist of two five (5) watt resistors whose series ohm value is equal to 13 ohms.

7) Ground J1 pin I to turn unit ON.

8) Using the voltmeter:

   a) Monitor the output voltage from J1 pin C for a minimum of 45 minutes. Voltage must be 24.0 ± 1.0 Vdc for at least 45 minutes after the load is applied.

   b) Check the output voltage from J1 pin H. Verify 5.0 ± 0.3 Vdc.

9) If the unit meets the standards in step 8, proceed to step 10. If the unit fails the standards in step 8, replace it.

10) Disconnect all loads (i.e. - J1 pin C and J1 pin H) and the ground from J1 pin I.

11) Repeat step 3. When the unit is fully charged, disconnect it from the test set-up and reinstall it.

B. Emergency Power Supply (S/N's 4636460, 4636463–4636652, less 4636633; and, 4692001 and up)

Emergency power for the standby attitude indicator is provided by a self contained power source. Required periodic maintenance is listed in 5-20-00 and 5-30-00. Checkout and test procedures and Instructions for Continued Airworthiness are provided in Mid-Continent Instruments Manual No. 9015762.

3. Horizontal Situation Indicator (HSI) or Electronic Horizontal Situation Indicator (EHSI)

NOTE: In airplanes equipped with either Avidyne Entegra or Garmin G1000/G1000 NXi, this function is provided by the Primary Flight Display (PFD).

The HSI or EHSI, which combines pictorial VOR/ILS Localizer indications with an electrical heading indicator, is the pilot's primary heading indicator. The gyro is not in the instrument case, but is located on the accessory shelf in the tail section of the airplane. The gyro is slaved to the earth's magnetic field through a flux detector (i.e. - flux gate compass) installed in the leading edge of the vertical stabilizer. Should the slaving mechanism become inoperative, the heading gyro can be deslaved through a switch on slave controlled unit mounted on the extreme upper left section of the pilot's instrument panel. Once deslaved, the heading gyro becomes free running, making it susceptible to gyroscopic precession. The compass control unit includes a spring-loaded to OFF switch marked as CW (clockwise) or CCW (counterclockwise), used to electrically drive the remote gyro to correct the heading indication for any precession. The system is protected through the 3 amp COMPASS circuit breaker located on the copilot's C/B panel.

A. Troubleshooting

See "Chart 2".

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 2
TROUBLESHOOTING ELECTRIC HEADING INDICATOR

Trouble	Cause	Remedy
Heading card "frozen" regardless if gyro is slaved or deslaved.	3 amp COMPASS circuit breaker open.	Reset breaker, if open.
	Faulty remote gyro unit.	Replace gyro unit.
	Break in wiring between remote gyro and instrument.	Check and repair broken wire(s).
	Faulty Indicator (Instrument).	Replace Indicator.
Heading card "frozen" or erratic when slaved, but works satisfactorily when deslaved.	Bad flux detector.	Replace flux detector.
	Faulty remote gyro unit.	Replace gyro unit.
Compass card erratic whether slaved or deslaved.	Faulty remote gyro unit.	Replace gyro unit.
	Faulty Indicator (Instrument).	Replace Indicator.
When deslaved, compass card will not respond to CW or CCW commands.	Faulty compass control unit.	Replace compass control control unit.
	Faulty remote gyro unit.	Replace gyro unit.

4. Air-Driven Heading Indicator (i.e., Directional Gyro)

NOTE: In airplanes equipped with either Avidyne Entegra or Garmin G1000/G1000 NXi, no air-driven heading indicator is installed. This function is provided by the Primary Flight Display (PFD).

The air-driven heading indicator is offered as an option for the copilot's navigation instrument package. The air-driven heading indicator is a flight instrument incorporating an air-driven gyro stabilized in the vertical plane. The gyro is rotated at high speed by lowering the pressure in the air-tight case and simultaneously allowing atmospheric air pressure to enter the instrument against the gyro buckets. Due to gyroscopic inertia, the spin axis continues to point in the same direction, even though the aircraft yaws to the right or left. This relative motion between the gyro and the instrument case is shown on the instrument dial, which is equipped with a 360° compass card. The card, when set to agree with the airplane's magnetic compass, provides a positive indication free from acceleration/deceleration and turning errors. However, the heading indicator has no sense of direction, and must be set to the magnetic compass. Since the magnetic compass is subject to errors due to magnetic fields, electric instruments, etc., the directional gyro is only accurate for the heading it has been set for. If the gyro is set on 270°, for instance, and the aircraft is turned to some other heading, there can be a large error between the gyro and the magnetic compass due to the error in compass compensation. This will appear as gyro precession. The gyro should frequently be checked for precession. Due to internal friction, span axis error, air turbulence and airflow, the gyro should be set at least every 15 minutes for accurate operation, whether it has drifted or not.

A. Troubleshooting

See "Chart 3".

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 3
TROUBLESHOOTING AIR-DRIVEN HEADING INDICATOR

Trouble	Cause	Remedy
Excess drift in either direction.	Setting error. Review	Review Heading Indicator above.
	Defective instrument.	Replace instrument.
	High or low vacuum. If vacuum is not correct, check for the following:	
	1. Relief valve improperly adjusted.	1. Adjust.
	2. Incorrect gauge reading.	2. Replace gauge.
	3. Pump failure.	3. Repair or replace.
	4. Vacuum line kinked or leaking.	4. Check and repair. Check for collapsed inner wall of hose.
Dial spins during turn.	Limits (55° bank) of gimbal exceeded.	Recage gyro in level flight.
Dial spins continuously.	Defective mechanism.	Replace.

5. Magnetic Heading Systems

A. S/N's 4636001 thru 4636247, less 4636160

The slaving magnetic compass installed in these airplanes was installed as part of the Bendix/King Autopilot/Flight Director installation and is serviced and adjusted per Bendix/King literature. See 22-10-00.

B. S/N's 4636160, and 4636248 thru 4636374

(1) Flux Detector

A flux detector installed in the leading edge of the vertical fin is used to provide heading data to the HSI/EHSI and, if installed, the ADAHRS. See "Figure 1" on page 34208.

(a) Removal

CAUTION: PERFORM "MAGNETIC HEADING COMPENSATION/CALIBRATION", BELOW, WHENEVER THE FLUX DETECTOR IS CHANGED.

1) Remove the dorsal fairing to expose the flux detector.
2) Disconnect the wiring harness(es) from the top of the flux detector to be removed.
3) Unscrew and remove the three brass screws and washers and remove the flux detector.

(b) Installation

CAUTION: THE FLUX DETECTOR IS SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Place the flux detector into position on its mounting bracket and secure with brass screws and washers (3 ea.)
2) Connect wiring harness to the connector on top of the flux detector.
3) Seal the connector backshell and base (see "Figure 1" on page 34208, View A) with DOW Corning 4 Electrical Insulating Compound.
4) Reinstall the dorsal fairing.

NOTE: Ensure correct hardware is used when reinstalling vertical fin dorsal fairing over flux detector.

(2) Magnetic Heading Compensation/Calibration

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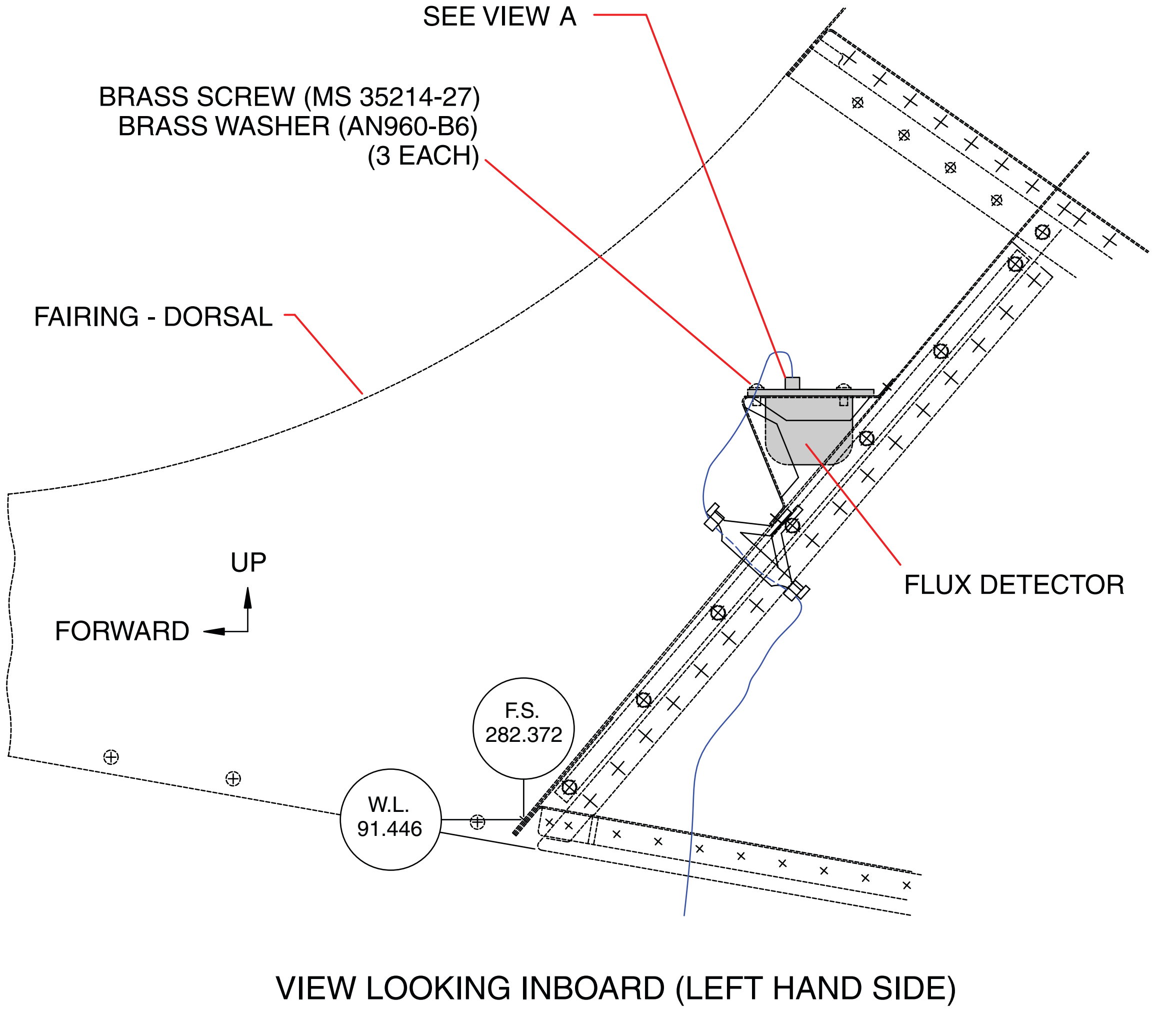
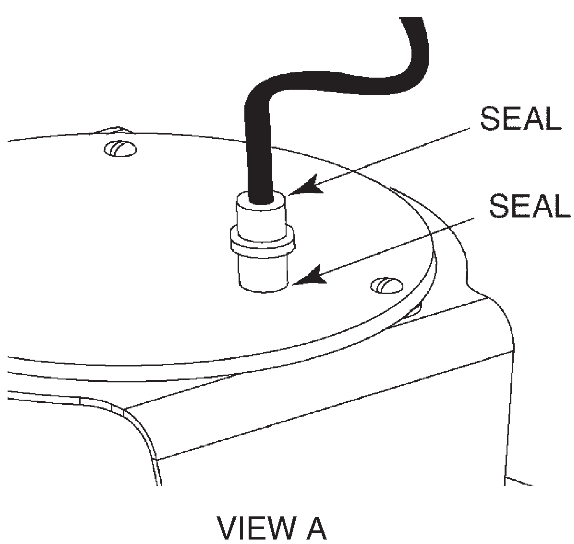
(a) Standard Installation

(PIR-PPS60191, Rev. New.)

Accuracy of the entire heading system is dependent on the location of the flux detector and proper calibration. Accuracies of plus or minus one degree are possible when care is taken during installation and calibration. To obtain such results:

- The flux detector must be positioned so that it points in the direction of aircraft flight; and,
- The north-south and east-west correctors must be adjusted to compensate for extraneous magnetic fields near the location of the flux detector.





Effectivity
4636160, and
4636248 thru 4636374

Flux Detector Installation
Figure 1

1) Required Equipment

Instrument Flight Research Corp. RF signal generator NAV-40IL or equivalent. Characteristics of the signal generator include: Frequency ranges 108 to 118 MHz, 117 to 136 Mhz, 328 to 336 MHz; +1/-0.01% accuracy, output level continuously adjustable from 1 .0uV to 0.1V into a 50 ohm load; 50 ohm output impedance; and internal adjustable or stepped VOR, LOC, and GS modulation. The RF signal generator should be portable and convenient for use while sitting in the aircraft cockpit.

2) Procedure

a) Apply power to the Model ST-180 HSI System. Allow at least three (3) minutes for the gyro to erect and synchronize.

b) Prior to actual alignment of the flux sensor, turn the aircraft to both north and east headings. Apply power to electrical equipment such as navigation and beacon lights and verify that the compass system is not affected.

c) Align the aircraft to an approximate magnetic north heading. On "Chart 4" on page 342010, record the actual magnetic heading and the HSI heading card reading.

d) Determine and record on "Chart 4" on page 342010 the deviation between the actual magnetic heading and the heading card heading. If the heading card reads high, the deviation is a plus.

e) Repeat steps "c)" and "d)" for east, south, and west headings. Record actual magnetic headings, heading card readings, and deviations on "Chart 4" on page 342010.

f) Plot deviations on the initial deviation graph in "Figure 2" on page 342010.

g) Realign the aircraft to north. Adjust the north-south corrector on the Slaving Panel, for one half of the difference between the north and south deviations. Record the new deviation for north and south on the initial deviation graph in "Figure 2" on page 342010.

h) Realign the aircraft to east. Adjust the east-west corrector on the Slaving Panel, for one half the difference between the east and west deviations. Record the new deviation for east and west on the initial deviation graph in "Figure 2" on page 342010.

i) If the pattern is not centered around zero, rotate the flux sensor clockwise to correct for minus deviations or counterclockwise for plus deviations. Plot final deviations on the final deviation graph in "Figure 2" on page 342010.

j) The deviations should now center around the zero reference line of the graph. If the error exceeds the specified system error limits (± 3 degrees), repeat the complete procedure.

(b) Optional Meggitt EFIS Installation

See "Magnetic Heading Compensation / Calibration" on page 342112.

C. S/N's 4636375 and up

(1) With Avidyne Entegra

See "Magnetic Heading Systems" on page 342213.

(2) With Garmin G1000

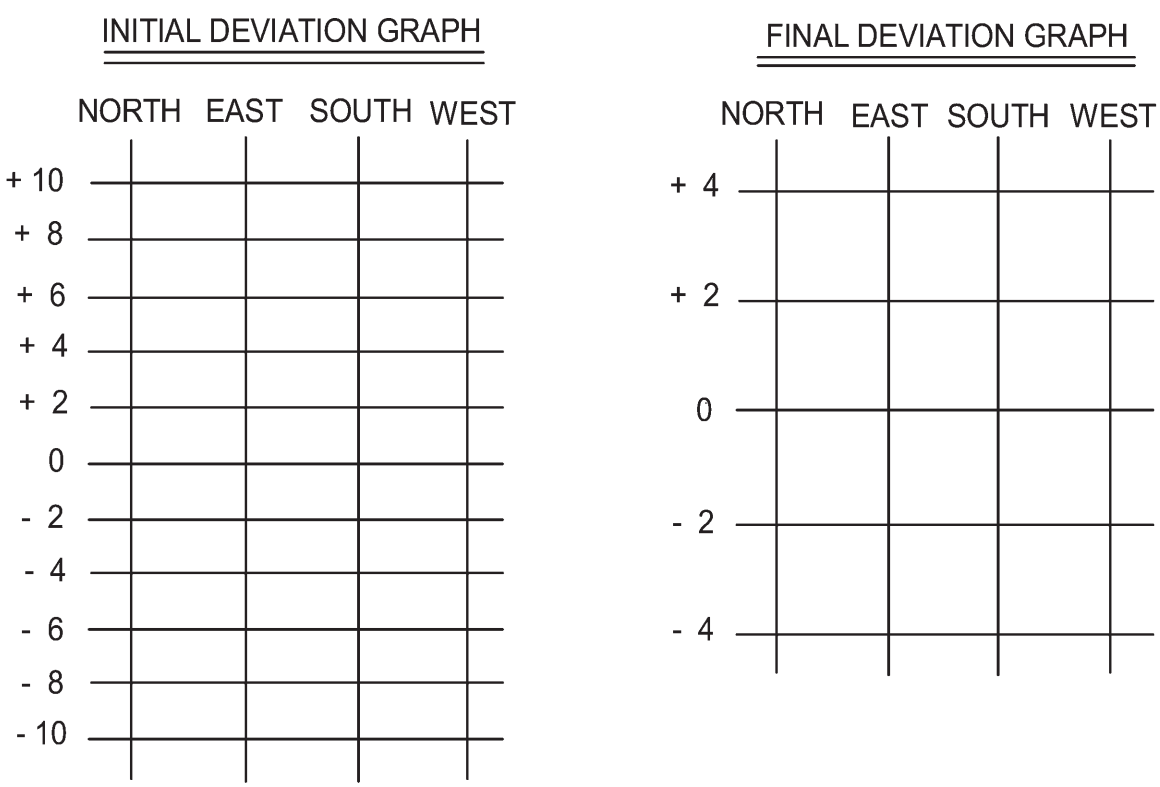
See "GRS 77 / GMU 44 Calibration Procedures" on page 342501190.

(3) With Garmin G1000 NXi

See "GRS 79 / GMU 44" on page 342502190.

CHART 4
DEVIATION CHART

Approx. Magnetic Heading	Actual Magnetic Heading	Heading Card Reading	Deviation
North			
East			
South			
West			



Effectivity
4636160, and
4636248 thru 4636374

ST-180 Deviation Graphs
Figure 2

D. Magnetic Compass

NOTE: A traditional fluid-filled magnetic compass is not installed in later PA-46-350P airplanes.

(1) Description

The magnetic compass is a self-contained instrument. This instrument has an individual light which is connected to the instrument lighting circuit. The compass correction card is located in the card holder mounted on the instrument.

Swing the compass as described under Adjustment, below, as follows:

(a) When the accuracy of the compass is suspected.

(b) After any cockpit modification or major replacement involving ferrous metal.

(c) Whenever a compass has been subjected to a shock; e.g., after a hard landing or turbulence.

(d) After aircraft has passed through a severe electrical storm.

(e) After lighting strike.

(f) Whenever a change is made to the electrical system.

(g) Whenever a change of cargo is likely to affect the compass.

(h) When aircraft's area of operation is changed to a different geographic location with a major change in magnetic deviation. (e.g., from Miami, Florida to Fairbanks, Alaska.)

(i) After aircraft has been parked on one heading for over a year.

(j) When flux valves / magnetometers are replaced.

(2) Troubleshooting

See "Chart 5" on page 342012.

(3) Adjustment

Before attempting to compensate compass, every effort should be made to place the aircraft in simulated flight conditions; check to see that the doors are closed, flaps in retracted position, engine running, throttle set at cruise position and aircraft in level flight attitude. Aircraft master switch, alternator switch, and all radio switches should be in the ON position. All other cockpit controlled electrical switches should be in the OFF position.

(a) Set adjustment screws of compensator on zero. Zero position of adjusting screws is when the dot of the screw is lined up with the dot of the frame.

(b) Head aircraft on a magnetic North heading. Adjust N-S adjustment screw until compass reads exactly North.

(c) Head aircraft on a magnetic East heading and do the same as step (2), adjusting E-W adjusting screw.

(d) Head aircraft on a magnetic South heading and note resulting South error. Adjust N-S adjusting screw until one-half of this error has been removed.

(e) Head aircraft on magnetic West and do same as step (4), adjusting E-W adjustment screw.

(f) Head aircraft in successive magnetic 30° degree headings and record compass readings on appropriate deviation card. Deviations must not exceed +10° on any heading.

CHART 5
TROUBLESHOOTING MAGNETIC COMPASS

Trouble	Cause	Remedy
Excessive card error.	Compass not properly compensated.	Compensate instrument.
	External magnetic interference.	Locate magnetic interference and eliminate if possible.
Excessive card oscillation.	Insufficient liquid.	Replace instrument.
Card sluggish.	Weak card magnet.	Replace instrument.
	Excessive pivot friction or broken jewel.	Replace instrument.
Liquid leakage.	Loose bezel screws.	Replace instrument.
	Broken cover glass.	Replace instrument.
	Defective sealing gaskets.	Replace instrument.
Discolored markings.	Age.	Replace instrument.
Defective light.	Burned out lamp or broken circuit.	Check and replace lamp. Check continuity of wiring.
Card sticks.	Altitude compensating diaphragm collapsed.	Replace instrument.
Card does not move when compensating screws are turned.	The gears that turn compensating magnets are stripped.	Replace instrument.
Compass swings erratically. when radio transmitter is keyed.	Normal.	

6. Turn Coordinator

NOTE: In airplanes equipped with either Avidyne Entegra or Garmin G1000/G1000 NXi, this function is performed solely by the Primary Flight Display (PFD).

Unlike the conventional turn and slip indicator, the electrically operated gyroscope in the turn coordinator is canted. However, like conventional turn and slip indicator, it works on the principal of precession. By canting the gyro, the instrument not only measures rate of turn, but also measures rate of roll. With this indicator, if the aircraft is rolled right and left rapidly, the indicator will move, measuring the rate at which the airplane is rolled by indicating a turn in the direction of the roll. If the aircraft is held in a bank, and rudder is applied (such as when slipping), the needle indicator will come back to neutral, indicating no turn.The slip/skid portion of the indicator is a ball sealed in a curved glass tube filled with damping fluid. In the previous example, It would indicate the airplane is slipping. By utilizing rudder and aileron to establish the airplane in a desired rate of turn will eventually establish the airplane in a coordinated turn at the desired rate.

A. Troubleshooting

See "Chart 6".

B. Removal and Installation

See 39-10-00, Face Mounted Instruments - Removal and Installation.

CHART 6
TROUBLESHOOTING TURN COORDINATOR

Trouble	Cause	Remedy
Instrument fails to respond when power is being applied to instrument.	Foreign matter lodged in instrument.	Replace instrument.
Incorrect sensitivity.	Out of calibration.	Replace instrument.
Incorrect turn rate.	Out of calibration.	Replace instrument.
Ball sticky.	Flat spot on ball.	Replace instrument.
Ball not in center when aircraft is correctly trimmed in wings level flight.	Instrument not level in panel.	Level instrument.
Instrument will not run.	No power to instrument.	Check appropriate circuit breaker on pilot's circuit breaker panel. Check circuit and repair.
	Instrument malfunction.	Replace instrument.

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MEGGITT ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

Meggitt Electronic Flight Instrument System (EFIS)

Optional in S/N's 4636349 thru 4636374.

A. Electronic Attitude Direction Indicator (EADI) [i.e., Meggitt MAGIC EFIS Primary Flight Display (PFD)]

The Primary Flight Display (PFD) is a microprocessor-based color, liquid crystal (LCD) display system. Information displayed includes Airspeed, Altitude, Attitude, Vertical Speed, Heading and Instrument Landing System data. The information is conveyed via predefined display formats. In addition, the PFD includes built-in test, performance and health monitoring functions and display brightness control. The PFD provides Gillam Code Altitude Output for the transponder (Mode C altitude).

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Descriptions of the operator controls and basic operating modes and screens are in the appropriate Pilot's Operating Handbook Supplement.

(1) Troubleshooting

See "Chart 1" on page 34212. See also Appendix 1.

(2) Basic Functional Test

CAUTION: IF, DURING THE FOLLOWING TEST, ANY COMPONENT DOES NOT PRODUCE THE EXPECTED RESULTS, THAT COMPONENT MUST NOT BE USED IN FLIGHT UNTIL THE PROBLEM IS CORRECTED.

(a) Before applying power, verify proper voltage inputs and ground connections as well as all system interfaces.

(b) Start the PFD in Test Mode by pressing and holding the Baro Pushbutton while applying power until the Initialization page is shown. Verify that the Initialization page displays 'MIRAGE PRIMARY FLIGHT DISPLAY' at the top of the screen and is replaced by the Configuration Data page after approximately five (5) seconds.

(c) Verify that the Configuration Data page Configuration Discrete Parity Status and Power On Self Test results both show results of Pass. If the result is FAIL, further diagnostic effort is required to determine if the failed indication is the result of improper installation, a faulty PFD, or both.

(d) Verify that the Configuration Data page displays the correct installation location (primary or secondary).

(e) Remove power from the PFD to terminate Test Mode. Re-apply power to the PFD and start the display in Mission Operational Mode.

(f) Verify that the PFD displays the Initialization page for approximately five (5) seconds prior to displaying the PFD Screen Format. If power has not been applied to the ADAHRS then the PFD screen format will be replaced by failure indications.

(g) Apply power to the Air Data Attitude Heading Reference System (ADAHRS) (i.e., AHRS).

(h) Verify that the PFD Screen Format is displayed (if the ADAHRS is performing initialization then the ADAHRS Initialization screen will replace the attitude sphere). If any of the failure indications are displayed refer to "Chart 1" on page 34212.

(i) Verify that the ADAHRS has completed initialization by observing that the attitude sphere replaces the ADAHRS Initialization screen. If the ADAHRS has not completed initialization within 180 seconds after application of power to the ADAHRS refer to "Chart 1" on page 34212.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING EADI (I.E., PFD)

Trouble	Cause	Remedy
No display visible at power-up.	+28 VDC power missing.	Check wiring & bus voltage.
	+28 VDC Return missing.	Check wiring.
	Failed PFD.	Replace PFD.
	EADI Display Down mode selected.	Check EADI Display Down switch.
Test mode visible at power-up.	Stuck Baro pushbutton.	Check Baro pushbutton.
	PFD failed self test.	Replace PFD.
Display brightness does not change when PFD brightness control operated.	28 VDC lighting bus not connected or not functional.	Check wiring and 28 VDC lighting bus.
	Brightness control not functional.	Check Brightness control.
	Failed PFD.	Replace PFD.
Barometric correction does not change when BARO Rotary control is rotated.	PFD in test mode.	Ensure PFD in mission operational mode.
	Failed PFD.	Replace PFD.
Barometric correction does not change to standard setting when Baro pushbutton is pressed.	PFD in test mode. Ensure	PFD in mission operational mode.
	Failed PFD.	Replace PFD.
Unable to enter test mode when Baro pushbutton is pressed.	Airspeed equal to or greater than 40 knots detected.	Ensure airspeed is less than 40 knots.
	Unit not powered off before Baro Pushbutton is pressed.	Ensure unit is powered off before Baro Pushbutton is pressed.
	Baro Pushbutton pressed and held for insufficient time.	Ensure Baro Pushbutton is pressed and held until initialization page displays.
	Failed PFD.	Replace PFD.
Test page does not change when Baro Rotary Control is rotated.	PFD in mission operational mode.	Ensure PFD in test mode.
	Failed PFD.	Replace PFD.
Power on self test status on Configuration Data page is indicated as FAIL.	Failed PFD.	Replace PFD.
Configuration discrete parity status on Configuration Data page is indicated as FAIL.	Parity input (Configuration Discrete 7) not wired correctly.	Check configuration discrete setting.
		Check wiring.
	Failed PFD.	Replace PFD.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING EADI (I.E., PFD)

Trouble	Cause	Remedy
Installation location on Configuration Data page incorrect.	Configuration discretes 0 and 1 not wired correctly.	Check configuration discrete setting.
		Check wiring.
	Failed PFD.	Replace PFD.
ADAHRS initialization screen is not removed from the display after 3 minutes have elapsed since ADAHRS power on.	ADAHRS not completing initialization.	Perform ADAHRS test.
Rad Alt indication and/or decision height annunciation are not displayed correctly.	Failed Radar Altimeter.	Perform Rad Alt test.
	Failed PFD.	Replace PFD.
Localizer/Glideslope and/or Backcourse/Localizer sensing indications not displayed correctly.	Failed VOR radio.	Perform VOR radio test.
	Failed PFD.	Replace PFD.
Marker beacon indications not displayed correctly.	Failed marker beacon system.	Perform marker beacon system test.
	Failed PFD.	Replace PFD.
Flight director bars not displayed correctly.	Failed Autopilot system.	Perform Autopilot system test.
	FD/AP switch not functional/wired correctly.	Check FD/AP switch and wiring.
	Failed PFD.	Replace PFD.
Full Field Red, Green, Blue, Black, and White pages are not displayed correctly.	Failed PFD.	Replace PFD.
ND Interface Data page shows valid data is not being transmitted.	Failed PFD.	Replace PFD.
ND Interface Data page shows valid data is not being received.	RS422 Receive A and B lines are not connected or are wired incorrectly.	Check wiring.
	Failed ND.	Replace ND.
ADAHRS Interface Data pages show valid data is not being received.	ARINC IN A and B lines are not connected or are wired incorrectly.	Check wiring.
	Failed ADAHRS.	Replace ADAHRS.
	Failed PFD.	Replace PFD.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING EADI (I.E., PFD)

Trouble	Cause	Remedy
Selecting the EADI Display Down switch to EADI Display Down does not remove the PFD display.	EADI Display Down switch not functional/wired correctly. Failed PFD.	Check EADI Display Down switch and wiring. Replace PFD.
PFD screen format not displayed on the PFD when the EADI Display Down switch selected to normal.	EADI Display Down switch not functional/wired correctly. Failed PFD.	Check EADI Display Down switch and wiring. Replace PFD.

(j) Apply power to the Navigation Display (ND) (i.e., EHSI).

(k) If installed, apply power to the Radar Altimeter. Perform a functional test on the Radar Altimeter system and verify that the PFD Rad Alt Indication and Decision Height Annunciation are displayed correctly.

(l) Apply power to the VOR radio. Perform a functional test on the VOR Radio system and verify that the PFD Localizer/Glideslope and Backcourse/Localizer Sensing Indications are displayed correctly.

(m) Apply power to the Marker Beacon system. Perform a functional test on the Marker Beacon system and verify that the PFD Marker Beacon indications are displayed correctly.

(n) Apply power to the Autopilot system. Place the Autopilot Master Switch in the Flight Director position and verify that the flight director bars are displayed correctly.

(o) Press the Baro pushbutton and verify that the barometric correction displays ‘STD’. Rotate the Baro Rotary Control clockwise and verify that the barometric correction increases. Rotate the Baro Rotary Control counter-clockwise and verify that the barometric correction decreases.

(p) On the overhead switch panel, decrease the Panel LCD’s dimmer switch and verify that the PFD brightness decreases.

(q) On the overhead switch panel, increase the Panel LCD’s dimmer switch and verify that the PFD brightness increases.

(r) Press the EADI Display Down switch. Verify that the PFD screen blanks and that the PFD screen format is displayed on the Navigation Display (ND) (i.e., EHSI) .

(s) Press the EADI Display Down switch again. Verify that the PFD screen format is displayed on the PFD and that the PFD screen format is removed from the ND.

(3) Removal and Installation

See 39-10-00.

B. Electronic Horizontal Situation Indicator (EHSI) [i.e., Meggitt MAGIC EFIS Navigation Display (ND)]

The Navigation Display (ND) is, like the Meggitt Primary Flight Display (PFD), a microprocessor-based color, liquid crystal (LCD) display system. Information displayed includes Navigation Data, Airspeed, Altitude, Attitude, Vertical Speed, Heading and Instrument Landing System data. The information is displayed via a series of predefined display formats (HSI, ARC, MAP). In addition, the ND includes built-in test, performance and health monitoring functions and display brightness control.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Descriptions of the operator controls and basic operating modes and screens are in the appropriate Pilot's Operating Handbook Supplement.

(1) Troubleshooting

See "Chart 2" on page 34216. See also Appendix 1.

(2) Basic Functional Test

The following test procedure should be performed with the aircraft in a suitable position to receive valid VOR/ILS/GPS/DME signals. If this is not possible, then use the appropriate test sets to simulate the signals as required.

CAUTION: IF, DURING THE FOLLOWING TEST, ANY COMPONENT DOES NOT PRODUCE THE EXPECTED RESULTS, THAT COMPONENT MUST NOT BE USED IN FLIGHT UNTIL THE PROBLEM IS CORRECTED.

(a) Before applying power, verify proper voltage inputs and ground connections as well as all system interfaces.

(b) Start the ND in Test Mode by pressing and holding the MNU Pushbutton while applying power until the Initialization page is shown. Verify that the Initialization page displays 'MERIDIAN NAVIGATION DISPLAY' at the top of the screen and is replaced by the Configuration Data page after approximately five (5) seconds.

(c) Verify that the Configuration Data page Configuration Discrete Parity Status and Power On Self Test results both show results of Pass. If the result is FAIL, further diagnostic effort is required to determine if the failed indication is the result of a fault in the wiring, a faulty ND, or both.

(d) Verify that the Configuration Data page displays the correct installation location (primary or secondary).

(e) Remove power from the ND to terminate Test Mode. Re-apply power to the ND and start the display in Mission Operational Mode.

(f) Verify that the ND displays the Initialization page for approximately five (5) seconds prior to displaying the HSI Screen Format. If power has not been applied to the ADAHRS then the HSI screen format will display the heading failure indication.

(g) Apply power to the Air Data Attitude Heading Reference System (ADAHRS) (i.e., AHRS).

(h) Apply power to the Primary Flight Display (PFD) (i.e., EADI).

(i) Verify that the HSI Screen Format is displayed and the installation specific defaults as shown in "Chart 3" on page 342110 apply. If any of the failure flags are displayed refer to "Chart 2" on page 34216.

(j) Enable the Mission Operational Mode/Rev Mode menu by pressing the bezel mounted 'MNU' button. Verify that the menu is displayed correctly.

CHART 2 (Sheet 1 of 3)
TROUBLESHOOTING EHSI (I.E., ND)

Trouble	Cause	Remedy
No display visible at power-up.	+28 VDC power missing.	Check wiring & bus voltage.
	+28 VDC Return missing.	Check wiring.
	Failed ND.	Replace ND.
Display powers-up in EADI Display Down mode.	EADI Display Down mode selected.	Check EADI Display Down switch.
	Failed ND.	Replace ND.
Test mode visible at power-up	Stuck MNU button	Check MNU button
	ND failed self test	Replace ND
Display brightness does not change when Panel LCD's dimmer switch operated	28 VDC lighting bus not connected or not functional	Check wiring and 28 VDC lighting bus
	Panel LCD's dimmer module not functional	Check/replace module
	Panel LCD's dimmer switch not functional	Check/replace switch.
	Failed ND	Replace ND
Menu does not appear when MNU button is pressed	Stuck MNU button	Check MNU button
	Failed ND	Replace ND
Menu cursor does not change when ↑ and/or ↓ buttons are pressed	Stuck ↑ and/or ↓ buttons	Check ↑ and ↓ buttons
	Failed ND	Replace ND
Unable to select menu options when SEL button is pressed	Stuck SEL button	Check SEL button
	Failed ND	Replace ND
Unable to enter Test mode when MNU button is pressed	Airspeed equal to or greater than 40 knots detected	Ensure airspeed is less than 40 knots
	Unit not powered off before MNU button is pressed	Ensure unit is powered off before MNU button is pressed
	MNU button pressed and held for insufficient time	Ensure MNU button is pressed and held until initialization page displays
	Failed ND	Replace ND
Test menu does not change when the ↑ and/or ↓ buttons are pressed	ND in mission operational mode	Ensure ND in test mode
	Stuck ↑ and/or ↓ buttons	Check ↑ and/or ↓ buttons
	Failed ND	Replace ND
Unable to select Test pages when the SEL button is pressed	Stuck SEL button	Check SEL button
	Failed ND	Replace ND

CHART 2 (Sheet 2 of 3)
TROUBLESHOOTING EHSI (I.E., ND)

Trouble	Cause	Remedy
Barometric setting does not change in EADI Display Down mode when the ↑ and/or ↓ buttons are pressed	Stuck ↑ and/or ↓ buttons	Check ↑ and ↓ buttons
	Failed ND	Replace ND
Selected CRS does not change when CRS knob rotated	Stuck CRS knob	Check CRS knob
	Failed ND	Replace ND
Selected HDG does not change when HDG knob rotated	Stuck HDG knob	Check HDG knob
	Failed ND	Replace ND
Selected CRS does not align with present heading when CRS knob pushed	Stuck CRS knob	Check CRS knob
	Failed ND	Replace ND
Selected HDG does not align with present heading when HDG knob pushed	Stuck HDG knob	Check HDG knob
	Failed ND	Replace ND
Failure flags present on display	Equipment supplying relevant data to ND failed	Check status of equipment supplying data to ND
	Failed ND	Check data on relevant ND test page
Power on self test status on Configuration Data page is indicated as FAIL	Failed ND	Replace ND
Configuration discrete parity status on Configuration Data page is indicated as FAIL	Parity input (Configuration Discrete 7) not wired correctly	Check configuration discrete setting Check wiring
	Failed ND	Replace ND
Installation location on Configuration Data page incorrect	Configuration discretes 0 and 1 not wired correctly	Check configuration discrete setting Check wiring
	Failed ND	Replace ND
PFD Interface Data Page shows valid data is not being received/transmitted	RS422 Receive and/or Transmit lines are not connected or wired correctly	Check wiring
	Failed ND	Replace ND
	Failed PFD	Replace PFD
ADAHRS Interface Data pages show valid data is not being received	ARINC IN A and B Bus 1 lines are not connected or are wired incorrectly	Check wiring
	Failed ADAHRS	Replace ADAHRS
	Failed ND	Replace ND

CHART 2 (Sheet 3 of 3)
TROUBLESHOOTING EHSI (I.E., ND)

Trouble	Cause	Remedy
NAV Interface Data Page shows valid data is not being Received	ARINC In A and B Bus 2 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
	Failed Nav radio	Replace Nav radio
GPS Interface Data Pages show valid data is not being Received	ARINC In A and B Bus 4A and/or 4B lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
	Failed GPS Receiver	Replace GPS Receiver
Cross Side Input Interface Data Page shows valid data is not being Received	ARINC In A and B Bus 3 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
	Failed cross-side ND	Test cross-side ND
Cross Side Output Interface Data Page shows valid data is not being Transmitted	ARINC Out A and B Bus 2 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
NAV/GPS Output Interface Data Page shows valid data is not being transmitted	ARINC Out A and B Bus 1 lines are not connected or are wired incorrectly	Check wiring
	Failed ND	Replace ND
Analog Interface Data Page shows valid data is not being Received or Transmitted	Failed ND	Replace ND
	Failed Autopilot Input	Perform Autopilot Test
	Failed DME Input	Perform DME Test
	Failed Audio Panel Input	Perform Audio Panel Test
	Failed Rad Alt Input	Perform Rad Alt Test
	Failed ADF Input	Perform ADF Test
	Failed Autopilot outputs	Check ND inputs
Discrete Interface Data Page shows valid data is not being Received or Transmitted	Failed ND	Replace ND
	Failed Autopilot Input	Perform Autopilot Test
	Failed DME Input	Perform DME Test
	Failed Rad Alt Input	Perform Rad Alt Test
	Failed Autopilot output	Check Nav inputs.

(k) Ensure that the Nav Radios are selected off. Use the bezel mounted “↑”, “↓” and “SEL” buttons to navigate through the menu, selecting RMI1 to display VOR1 and RMI2 to display VOR2. Verify that the RMI1 and RMI2 second level menus are displayed correctly and the RMI1 and RMI2 source annunciations match the selections.

(l) Verify that the RMI1, RMI2 and CDI source annunciations are amber in color. Verify that the associated frequency annunciations are replaced with amber dashes. Verify that both RMI pointers and the CDI pointer are removed from the display.

(m) Apply power to the Nav radios. Select valid VOR frequencies on the Nav Radios and verify that the RMI1, RMI2 and CDI source and frequency annunciations are displayed in the correct colors (see "Chart 4" on page 342110). Verify that the ND displayed frequencies match the Nav radio frequencies (i.e. RMI1 frequency matches Nav Radio 1 and RMI2 frequency matches Nav Radio 2. The CDI frequency should match the currently selected CDI source frequency). Verify that both RMI pointers and the CDI pointer are displayed correctly.

(n) Select valid ILS frequencies on the Nav Radios and verify that the RMI1, RMI2 and CDI source annunciations display VOR1, VOR2 and ILS1/ILS2 (dependent on current selected CDI source) respectively in the correct colors (see "Chart 4" on page 342110). Verify that the ND displayed frequencies match the Nav radio frequencies (i.e. RMI1 frequency matches Nav Radio 1 and RMI2 frequency matches Nav Radio 2. The CDI frequency should match the currently selected CDI source frequency). Verify that both RMI pointers are removed and the CDI pointer is displayed correctly.

(o) Apply power to the GPS Receivers. Use the bezel mounted “↑”, “↓” and “SEL” buttons to navigate through the menu, selecting RMI1 and RMI2 to display GPS and CDI (verify that the CDI second level menu is displayed correctly) to display GPS (GPS1 or GPS2 will be available dependent on GPS source selection). Verify that the RMI1, RMI2 and CDI source annunciations reflect the selections.

(p) Refer to the GPS Installation Manual and display the self test page on the on-side GPS. Verify that the applicable ND parameters display the correct self test values as listed in the GPS Installation Manual during the display of the self test page.

(q) Verify that the GPS source annunciation displays the currently selected GPS source in white. Use the bezel mounted “↑”, “↓” and “SEL” buttons to navigate through the menu, selecting the GPS source to the off-side GPS. Verify that the GPS second level menu is displayed correctly. Verify that the GPS source annunciation displays the currently selected GPS source in white.

(r) Refer to the GPS Installation Manual and display the self test page on the off-side GPS. Verify that the applicable ND parameters display the correct self test values as listed in the GPS Installation Manual during the display of the self test page.

(s) Press the EADI Display Down switch to select EADI Display Down mode and verify that the ND displays the PFD (i.e., EADI) screen format. Verify that the PFD screen blanks.

(t) If installed, apply power to the Radar Altimeter. Perform a functional test on the Rad Alt system and verify that the ND Rad Alt Indication and Decision Height Annunciation are displayed correctly.

(u) Perform a functional test on the VOR Radio system and verify that the ND Localizer / Glideslope and Backcourse / Localizer Sensing indications are displayed correctly.

(v) Apply power to the Marker Beacon system. Perform a functional test on the Marker Beacon system and verify that the ND Marker Beacon indications are displayed correctly.

(w) Apply power to the Autopilot system. Place the Autopilot Master Switch in the Flight Director position and verify that the flight director bars are displayed correctly.

CHART 3
NAVIGATION DISPLAY (ND) DEFAULT SETTINGS

Screen Format	HSI
Element	**Setting**
CDI	VOR1
RMI1	OFF
RMI2	OFF
RANGE	2.5NM
GPS SEL	GPS1
APORTS	Disabled
NAVAIDS	Disabled
WPTS	Disabled
NDB	Disabled
Selected Heading	Last Known Setting
Selected Course / Desired Track	Last Known Setting
Barometric Setting	Last Known Setting

CHART 4
NAVIGATION DISPLAY (ND) COLOR CODING

	Display Parameter	
Color	**Mission Operational Mode**	**EADI Display Down Mode**
White	A/C Symbol, Scales, Range Rings, Digital Readouts, RMI1 Pointer and Annunciation	Scales, Digital Readouts
Grey	N/A	Scale Tapes
A/C Symbol	N/A	Aircraft Symbol, Lubber Line
Amber	Failures and Annunciations	Stall Warning, Failures and Annunciations
Red	N/A	Airframe Limits
Sky	N/A	Sky
Ground	N/A	Ground and Radar Altimeter ground reference
Green	N/A	Baro Correction
Magenta	Deviation Bar, Sel HDG/CRS Bugs, CDI Pointer and Annunciation	Flight Director Command Bars and Deviation Pointers
Cyan	RMI2 Pointer and Annunciation	N/A

(x) Press the bezel mounted "↑" button once and verify that the barometric correction increases by 0.01in.Hg/1hPa. Press and hold the bezel mounted "↑" button and verify that the barometric correction continuously increases in increments of 0.01in.Hg/1hPa until the button is released or the maximum barometric correction is reached.

(y) Press the bezel mounted "↓" button once and verify that the barometric correction decreases by 0.01 in·Hg/1hPa. Press and hold the bezel mounted "↓" button and verify that the barometric correction continuously decreases in increments of 0.01 in·Hg/1hPa until the button is released or the minimum barometric correction is reached.

(z) Press the EADI Display Down switch to shift back to normal mode and verify that the HSI screen format is displayed on the ND. Verify that the PFD screen format displays on the PFD.

(aa) Use the bezel mounted "↑", "↓" and "SEL" buttons to navigate through the menu, selecting the Arc screen format. Verify that the Arc screen format is displayed correctly.

(ab) Use the bezel mounted "↑", "↓" and "SEL" buttons to navigate through the menu, selecting the Map screen format. Verify that the Map screen format is displayed correctly.

(ac) Use the bezel mounted "↑", "↓" and "SEL" buttons to navigate through the menu, selecting the HSI screen format. Rotate the CRS Rotary Control clockwise and verify that the selected CRS pointer rotates clockwise around the compass card and the selected course digital readout increases and mirrors the pointer heading. Rotate the CRS Rotary Control anti-clockwise and verify that the selected CRS pointer rotates anti-clockwise around the compass card and the selected course digital readout decreases and mirrors the pointer heading. Push the CRS Pushbutton and verify that the selected course pointer and digital readout align to the current aircraft heading.

(ad) Rotate the HDG Rotary Control clockwise and verify that the selected HDG bug rotates clockwise around the compass card and the selected heading digital readout increases and mirrors the bug heading. Rotate the HDG Rotary Control anti-clockwise and verify that the selected HDG bug rotates anti-clockwise around the compass card and the selected heading digital readout decreases and mirrors the bug heading. Push the HDG Pushbutton and verify that the selected heading bug and digital readout align to the current aircraft heading.

(ae) On the overhead switch panel, decrease the Panel LCD's dimmer switch and verify that the ND brightness decreases.

(af) On the overhead switch panel, increase the Panel LCD's dimmer switch and verify that the ND brightness increases.

(3) Removal and Installation

See 39-10-00.

C. Air Data & Attitude Heading Reference System (ADAHRS)

The ADAHRS unit is located on the aft equipment shelf behind the cabin rear closeout panel.

NOTE: When installing the ADAHRS unit; first, level the airplane per 8-20-00; and then, use NAS1149FN816P washers as required to level the ADAHRS unit.

(1) Ground Test Procedure

NOTE: For Meggitt-equipped airplanes, if any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per this Ground Test Procedure.

This test procedure (included as Appendix 1) verifies the integrity and accuracy of the air data and pitot-static systems and shows that these systems comply with the applicable airworthiness regulations. Perform this procedure:

(a) Each 24-months,

(b) To proof test the static system each time any connections in static system are opened for maintenance, or

(c) Following installation or maintenance on the ADAHRS unit, the primary flight display (PFD), or the transponder Mode C interface.

(2) Magnetic Heading Compensation / Calibration

(PIR-PPS 60189-1, Rev. New.)

The Air Data & Attitude Heading Reference System (ADAHRS) unit mounted on the aft equipment shelf provides the other components of the Meggitt Magic EFIS system with magnetic heading data derived from the flux detector installed in the leading edge of the vertical fin. The ADAHRS unit should be swung (checked for changes in deviation) whenever the ADAHRS unit, flux detector, or either cabin recirculation blower is changed, and at least once a year.

Use the following procedure to compensate and calibrate the magnetic heading indication portion of the Meggitt Magic EFIS installation:

(a) Setup

1) Level the airplane within ± 1 degree laterally and longitudinally at the top surface of the seat rails.

2) Configure the airplane as follows:

External Power Applied to Aircraft	Avionics - ON
Instrument Panel Lights - Full Bright	LCD Displays - Full Bright
Day/Night Switch - Day	Navigation Lights - ON
Strobe Lights - ON	Pitot Heat - OFF
Windshield Heat - OFF	Air conditioning - OFF
Cabin Blower Fan - Low	Vent Blower - OFF

(b) ADAHRS Unit Alignment Procedure

1) Place the EHSI in the “TEST MODE” by applying power with the “MNU” button held in the depressed position until the initialization page is displayed.

2) Select ADAHRS test page 2 from the EHSI menu selections.

3) Allow the ADAHRS to complete the initialization period and display an attitude sphere on the EADI.

4) Verify that the pitch and roll errors, as displayed on the EHSI test page, are less than ± 0.5 degrees in either axis.

5) If required for ADAHRS leveling, install shims under the ADAHRS mounting feet and re-secure the unit to the mounting structure.

6) Repeat the alignment procedure until the minimum acceptable criteria is met.

(c) Compass System Calibration Procedure

1) Flux Detector Indexing

CAUTION: THE TEST CABLE P/N 101453-002 IS REQUIRED EQUIPMENT. NO OTHER TEST CABLE IS AUTHORIZED. SEE "FIGURE 2" ON PAGE 342116.

a) Install the ADAHRS test cable and verify that the ADAHRS test cable calibration switch is in the “NORMAL” (open) position.

b) Place the EHSI in the “TEST MODE” by applying power with the “MNU” button held in the depressed position until the initialization page is displayed.

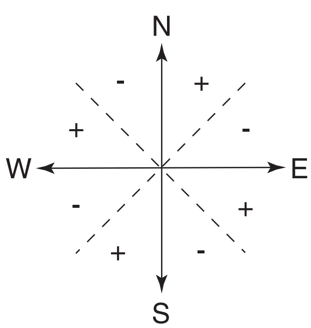
c) Select the ADAHRS interface test page 2 from the Navigation Display (ND) menu selection. The ADAHRS must be in the “INITIALIZATION” mode for the indexing procedure.

d) Place the ADAHRS test cable calibration switch in the “CAL” (closed) position. Ensure that the heading system flag comes into view on the EADI, replacing the heading scale tape and the digital readout.

NOTE: If no calibration has ever been entered into the ADAHRS memory, a heading flag will be in view at the time of the unit initial power-up.

e) Position the aircraft on the NORTH heading of the compass rose and record the indicated “MAGHDG” as displayed on the ND. Record the heading indication in "Chart 5" on page 342114. (This is the raw data output from the flux detector). Repeat this step for EAST, SOUTH, and WEST headings.

f) After recording the raw heading information, calculate and record the index heading errors in the Chart mentioned above. Refer to "Figure 1" to calculate the errors. The following logic applies to calculating heading errors:



Calculating Heading Error
Figure 1

For N, if indication is +, then error is +
For N, if indication is -, then error is -
For E, if indication is more than 90, then error is +
For E, if indication is less than 90, then error is -
For S, if indication is - and less than 180, (i.e. -179) then error is +
For S, if indication is + and less than 180, (i.e. +179) then error is -
For W, if indication is - and less than 90, (i.e. -89) then error is +
For W, if indication is - and greater than 90, (i.e. -91) then error is -

Enter the errors for the four cardinal headings in "Chart 5" on page 342114. Add the errors and enter the sum in the “TOTAL ERROR” block. Divide the Total Error by 4 and enter the result in the “INDEX ERROR” block. This result is the “Index Error” of the flux detector.

g) If the index error is greater than one (1) degree, loosen the flux detector and adjust out the index error by rotating the flux detector.

h) Repeat steps (e) through (g) until the index error is less than one (1) degree.

NOTE: Upon completion of the flux detector indexing procedure, apply torque stripe to each of the flux detector mounting screws.

i) This completes flux detector indexing, proceed to Heading System Calibration / Compensation, below.

CHART 5
INDEXING DATA

Aircraft S/N:	Registration No.:	
ND Error (±)	**ND Heading**	**Compass Rose**
		N
		E
		S
		W
	Total Error (N+E+S+W)	
	Index Error (Total Error ÷ 4)	

CHART 6
HEADING SYSTEM CALIBRATION

ND Heading	Compass Rose
	N
	30
	60
	E
	120
	150
	S
	210
	240
	W
	300
	330

2) Heading System Calibration / Compensation

a) Heading system calibration / compensation should be performed upon completion of flux detector indexing, above, and using the same aircraft configuration specified in setup, above.

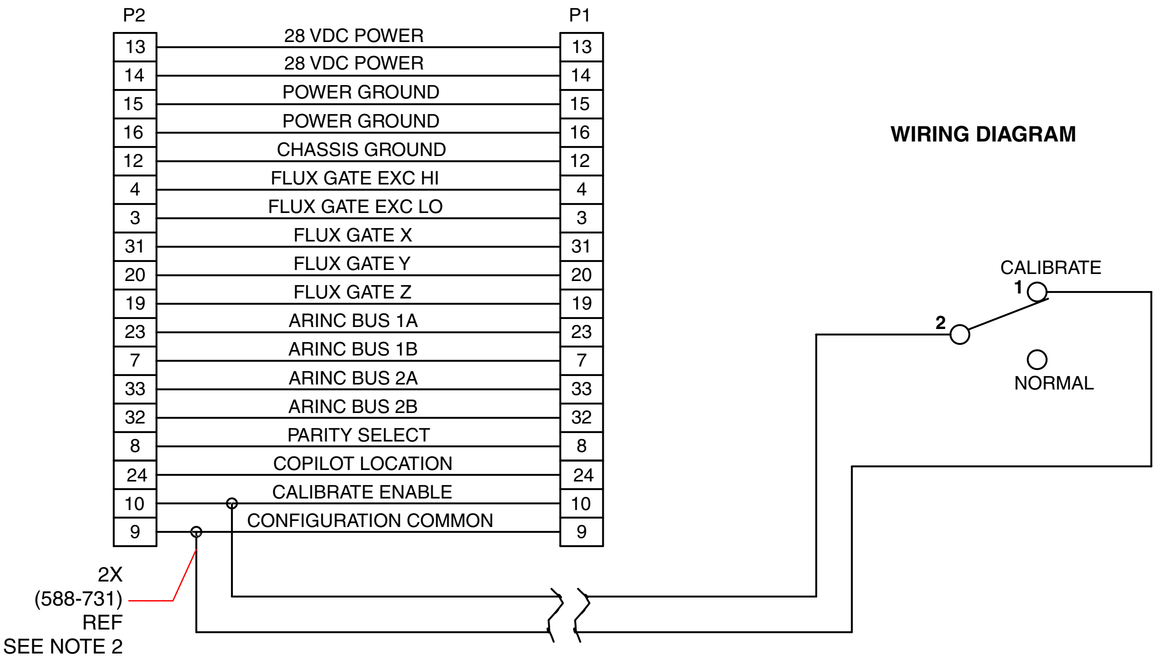
b) Place the Navigation Display (ND) in the “NORMAL” or mission operational mode (normal power-up).

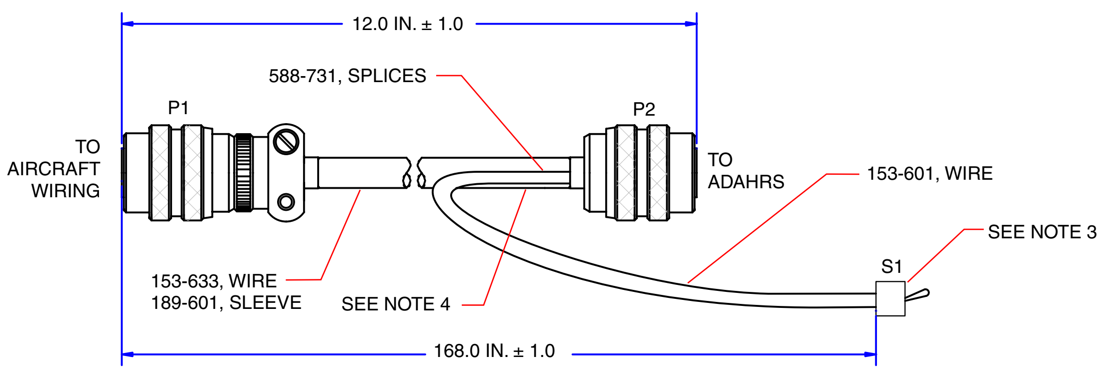
c) With the ADAHRS test cable installed, and the calibration switch in the “CALIBRATE” (closed) position, slowly taxi the aircraft in two (2) complete 360-degree turns. The direction of the turn does not matter; however, a minimum time of one (1) minute for each 360-degree turn (two (2) minutes minimum for the full 720 degrees) is required for the calibration process. The slow turn rate permits more data point samples for constructing the compensation curve. Make sure that the turns are continued until the heading flag is replaced by a normal heading indication.

d) Upon completing the two turns, the magnetic heading will automatically be displayed on both the EADI and EHSI. Place the ADAHRS test cable calibration switch in the “NORMAL” (open) position. Allow the compass system to stabilize for one (1) minute.

e) With the ADAHRS test cable calibration switch in the NORMAL (open) position, perform a complete heading system “swing”. A settling time of two (2) minutes must be used at each heading to ensure heading stabilization. After two (2) minutes of settling, record the indicated heading against the known compass rose heading for all twelve compass rose points in "Chart 6" on page 342114. The heading system error should not exceed ± 2 degrees on any compass point.

NOTES:
1. COVER 153-633 WIRE WITH WHITE SHRINK SLEEVE 189-601.
2. USE RAYCHEM SPLICES, 588-731 TO CONNECT TO SWITCH TO WIRES SHOWN IN WIRING DIAGRAM.
3. LABEL SWITCH CALIBRATION POSITION WITH "CAL".
4. LABEL ASSEMBLY AS FOLLOWS "FOR CALIBRATION PURPOSE ONLY,"NOT APPROVED FOR PERMANENT INSTALLATION.





	ITEM NO.	ITEM NAME	CODE NO. SPEC CONTROL DWG	MATERIAL	STOCK SIZE	MATERIAL SPECIFICATION	HEAT TREAT	FINISH SPECIFICATION	E-ITEM GROUP	ITEM TYPE	ITEM GROUP	NOTE	ZONE	POS. NO.
1	602-513	SWITCH, SPDT				MTL-106D								9
1	599-698	CONNECTOR, RECEPTACLE				MS27473T14B-35S								8
1	599-697	CONNECTOR, STRAIGHT PLUG				MS27508E14B35P								7
2	588-731	SPLICES				PS10025-16-2								6
A/R	189-601	SLEEVE, PTFE SHRINK, WHT				M23053/12-203-3								5
A/R	153-633	WIRE, PTFE, 22AWG WHT				M22759/11-22-9								4
A/R	153-601	WIRE, CABLE-EL				PMS-E0017-22-24								3
														2
	-002	HARNESS ASSEMBLY-ADAHRS CALIBRATION							3400	M	1530			1

INSTL, ASSY, SUB ASSY, OR KIT	REVISION						
						A	REV
						DC	BY
						6/16/03	DATE
							APPD
							REV
							BY
							DATE
							APPD

DRAFTSMAN	T TAYLOR	5/28/02
DESIGN ENGR		
CHECKER		
PROGRAM MGR		

SCALE | LAST SECTION OR VIEW
WT. | DRAWING SIZE D
TOL. .XXX = +/- .010 .XX = +/- .030 .X = +/- .10

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THE NEW PIPER AIRCRAFT, INC.
VERO BEACH, FLORIDA

MANUFACTURING PRACTICES - SEE PIPER REPORT VB-9
MANUFACTURING
MATERIALS

ADAHRS CALIBRATION
HARNESS ASSEMBLY

REF. LAYOUT NO. (S)

SH 1 OF 1 | 101453 | A

ADAHRS Calibration Harness Assembly
Figure 2

AVIDYNE ENTEGRA INTEGRATED FLIGHT DISPLAY SYSTEM (IFDS)

Avidyne Entegra Integrated Flight Display System (IFDS)

The Avidyne FlightMax Entegra Integrated Flight Display System (IFDS) is installed as standard equipment in PA-46-350P S/N's 4636375 thru 4636459, 4636461 thru 4636462, and 4636481; and PA-46R-350T S/N's 4692001 thru 4692133, 4692141, 4692149 and 4692153.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.

See 28-40-00 and 34-10-00 for more information.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 341014.

A. Description

See "Figure 1" on page 342210.

This system uses two large 10.4-inch diagonal, high-resolution, sunlight-readable full color displays (PFD and MFD), to provide primary flight and engine information as well as a wide variety of other data. Standard primary flight instruments (i.e., airspeed, electric attitude indicator, and altimeter) provide redundancy.

The installation consists of the following components: Primary Flight Display (PFD), Multifunction Display (MFD), Data Acquisition Unit (DAU), and associated sensors, and Magnetometer/OAT Sensor Assembly.

B. Maintenance

The Instructions for Continued Airworthiness (ICA) published by Avidyne provide the necessary information for maintaining this system as installed in Piper airplanes, except as noted below.

(1) Primary Flight Display (PFD) (PIR-PPS60227, Rev. B. / PPS60227-1, Rev. New. / PPS60232, Rev. B.)

Use 700-00006-0XX PFD & 700-00011-0XX Mag/OAT ICA, Avidyne Document No. AVPFD-174, Revision 03, or later, with the following exceptions:

NOTE: Before attempting to set-up the PFD, ensure the GNS 430(s) / GNS 430W(s) have been configured per the "Post Installation Setup Procedure" on page 34501 under COM/NAV/GPS in 34-50-00.

NOTE: In S/N's 4636349 and up, before attempting to set-up the PFD, ensure the GTX 330 is configured per the "Post Installation Setup Procedure" on page 34508 under Transponder, S/N's 4636349 and up.

NOTE: Whenever the PFD is replaced, perform the setup / calibration procedures specified in Avidyne Document No. AVPFD-174 as modified below.

NOTE: The following PFD software part numbers were factory installed in these airplanes: (the earliest of these versions may have been upgraded in the field)

530-00177-000 (i.e., Release 6.1 and earlier)

530-00183-000 (i.e., Release 6.2 and later)

530-00194-000 (i.e., Release 7 or later)

(a) In para 5.1, for airplanes equipped with PFD Software 530-00194-000 or later, the “ADU Calibration” referenced in conjunction with the bi-annual altimeter check can be performed by following the prompts from the ADU Calibration page.

(b) In para 6, Troubleshooting Information, in the chart where it says "OAT (Optional)," cross out "optional." The OAT is standard in the Piper installation.

(c) In para 7.2, "Primary Flight Display Installation," and Figure 7, the standard Avidyne installation describes alignment and mating pins and retaining clips on the sides of the PFD. For the Piper installation, however, there is only a single alignment pin on the top of the PFD engaging a slot in the upper rear cross bracket.

(d) Replace paragraphs 7.3 and 7.4 with "Magnetic Heading Systems" on page 342213.

(e) In para 7.5.1, apply power by turning the Battery Master Switch ON or by applying external power and turning the Radio Master Switch ON.

(f) When the countdown is finished and the system setup page appears, select the "System Info" tab and verify checksums have no "FAILED" or "PENDING" messages.

(g) Add the following to the end of para 7.5.2.1:

With PA-46-350P selected, verify the following:

PFD Serial Number:	XXXXXXXXXXX
S/W Version ID:	XXXXXXXXXXX
Aircraft Make:	Piper
Model:	PA-46-350P
Pitch Offset:	0.0
Avionics:	
GPS 1:	Garmin GNS 430/530
GPS 2:	Garmin GNS 430/530
VHF 1:	Garmin GNS 430/530
VHF 2:	Garmin GNS 430/530
Autopilot:	S-TEC System 55X
Engine DAU1:	"Moritz DAU"
Engine DAU2:	Not Installed
ADF:	Not Installed
Radar Altimeter:	Not Installed
Copilot PFD:	Installed (PA-46-350P) / Not Installed (PA-46R-350T)
Dimming Bus:	28 VDC

NOTE: If GPS 1, GPS 2, VHF 1, or VHF 2 do not display the correct Garmin unit, proceed to para 7.6 (or for airplanes with PFD Software 530-00183-000 or later, para 7.7) in Avidyne Document No. AVPFD-174 as modified by step "(i)" or "(j)", below, and update the display. Then repeat step "(g)".

(h) Replace para 7.5.3, Barometric Unit Setting, with the following:

With the "Display" tab selected, make the following selections:

1) Press LSK L1, labeled "Trim Ann", as required until "Show" is displayed.
2) Press LSK L2, labeled "A/P Annun", as required until "Show" is displayed.
3) Press LSK L3, labeled "V-Speeds", as required until "Show" is displayed.
4) Press LSK R1, labeled "Baro Unit", as required until "in.Hg", or "mb" if foreign, is displayed.
5) Press LSK R2, labeled "Horiz Marks", as required until "Show" is displayed.
6) Press LSK R3, labeled "ARS", as required until "Delta" is displayed.

(i) In para 7.6, add the following:

Verify that the Avionics Page Display indicates:

Aircraft Make:	Piper PA46
Model:	PA-46-350P (Mirage)
Sensor Suite:	
GPS 1:	GARMIN 430/530 on ARINC 1
GPS 2:	GARMIN 430/530 on ARINC 3
VHF 1:	GARMIN 430/530 on ARINC 2
VHF 2:	GARMIN 430/530 on ARINC 4
AutoPilot:	STEC 55X
Engine DAU:	INSTALLED
ADF:	NOT INSTALLED (even if an ADF is installed)
Copilot PFD:	INSTALLED

(j) In para 7.7, add the following:

Verify that the Avionics Page Display indicates:

GPS 1:	Garmin GNS-430/530
Inputs:	ARINC Rx1
Outputs:	ARINC Tx0
GPS 2:	Garmin GNS-430/530
Inputs:	ARINC Rx3
Outputs:	ARINC Tx0
VHF 1:	Garmin GNS-430/530
Inputs:	ARINC Rx2
VHF 2:	Garmin GNS-430-530
Inputs:	ARINC Rx4
Autopilot:	S-Tec System 55X
Attitude Rate Source:	External
Attitude Selector:	Emulating STEC ST-360
Annunciators:	Emulating STEC ST-645
Horizontal NAS Data:	Emulating Bendix/King KCS 55A
Flight Director:	Analog
GPS Roll Steering:	Retransmitting on ARINC Tx0
Engine DAU 1:	Moritz DAU
Engine DAU 2:	Not Installed
ADF:	Not Installed
Radar Altimeter:	Not Installed
Copilot PFD:	Installed (PA-46-350P) / Not Installed (PA-46R-350T)
Dimming Bus:	28 V DC

(k) In para 7.7.1, in the “Main RS232 Configuration Page” table, for CHNL 3 under GNS- 430 No 2, both Input and Output should read “Crossfill” instead of “Off.”

(l) In para 7.7.2, below the heading “Autopilot Setup,” insert the following:

“If the PFD is replaced, Magnetometer calibration must be completed before continuing with the following steps.”

(m) Replace para 7.7.3.1, "Calibrating the Magnetometer," with "Calibration" on page 342213 under "Magnetic Heading Systems".

(2) Multifunction Display (MFD)

(PIR-PPS60226, Rev. A. / PPS60231, Rev. B.)

Use 700-00004-0XX-() Multifunction Display ICA, Avidyne Document No. AVMFD-167, Revision 06, or later, with the following exceptions:

NOTE: Whenever the MFD is replaced, perform the System Setup and Checkout in paragraph 7.3, Avidyne Document No. AVMFD-167, as modified herein.

(a) In para 2, items 7 and 8 are standard in the Piper installation.

(b) In para 7.3.1, Maintenance Mode Access, replace 3 and 4 with the following:

1) At the prompt "Press any bezel key to continue", press any Line Select Key (LSK). This places the screen at the exceedance page. Press OK to acknowledge exceedances and to continue to the Initial Usable Fuel Page. Depress the "Fuel Done" LSK R4.

2) Rotate the right outer concentric knob clockwise until the AUX page is displayed.

(c) Replace 7.3.2.1, GAMA 429 Graphics Setup (EX5000 only) with the following:

1) From the maintenance page, depress the LSK L1 to access the GPS setup page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration; configure the GPS Setup page for the following conditions:

Receiver 1:	GAMA 429 FORMAT	**Receiver 2:**	GAMA 429 FORMAT
Port:	ARINC 429 RX 1	**Port:**	ARINC 429 RX 2
Speed:	LOW	**Speed:**	LOW

3) Press SAVE button.

(d) Replace 7.3.4, Lightning Sensor Interface and Setup, with the following:

1) From the maintenance page, depress LSK (L3) to access the Lightning Set-up page. In PA-46R-350T only, select "TWX-670" set-up, LSK (L1).

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration. Configure the Lightning Setup page for the following conditions:

	PA-46-350P	**PA-46R-350T**
Sensor:	WX-500	AVIDYNE TWX 670
Operating Mode:	Weather	Weather
Port:	RS232 3 (Default)	RS232 3 (Default)
Stabilization type:	Use Map Heading/Track	N/A
Enable lightning ahead:	☑	N/A
Antenna on top:	☐	N/A
Use leading source:	N/A	☑
Use position:	N/A	☑
Enable audio output:	N/A	☑

3) Press SAVE button.

(e) In para 7.3.5 and subordinate, use the following settings where appropriate:

		PA-46-350P	**PA-46R-350T** TAS 610 installed	**PA-46R-350T** TAS 610 not installed
1)	Sensor:	TAS	RS232 DEVICES	TIS-G
2)	Port:	ARINC 429 RX3	RS232 2	ARINC 429 RX3
3)	Type:	Bendix / King	TCAD 9900BX	

(f) In PA-46-350P only, replace 7.6.3.1, TAWS Setup, with the following:

1) From the maintenance page, depress LSK (L5) to access the TAWS Set-up page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration. Configure the TAWS Setup page for the following conditions:

Sensor:	Honeywell EGPWS
ARINC 429 Port:	ARINC 429 TX 2
ARINC 453 Port:	ARINC 453 2 – (TAWS Default)
Altitude Annunciation on:	☑

3) Press SAVE button.

(g) In PA-46-350P only, replace 7.3.7, Radar Sensor; and 7.3.12.7, Radar Setup, with the following:

1) Setup

If configuring a replacement MFD, start with step (a). In all other cases start with step (f).

a) From the maintenance page, depress LSK (R3) to access the RADAR Set-up page.

b) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration.

c) From the RADAR Set-up Page select the following:

Sensor:	AlliedSignal ART-2000
ARINC 429 Port:	ARINC 429 Tx 1
ARINC 453 Port:	ARINC 453 1

d) Press SAVE. (After pressing SAVE, MFD goes to maintenance Setup Page)

e) From the Maintenance Setup Page, press Restart System.

f) Re-enter Maintenance Setup Page, depress LSK R3 to access the RADAR Setup Page.

g) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration. Configure the Radar Setup page for the following conditions:

Radar Sensor:	AlliedSignal ART-2000
Park Position:	Centered
Beam Width:	7.0 °
Beam Height:	7.0 °

All boxes CHECKED ☑, except “Disable Stabilization” is NOT CHECKED ☐.
Press SAVE. (After pressing save, MFD goes to Maintenance Setup Page)

2) Calibration

The R/T Configuration Module must be configured using the Allied Signal KPA 900 Configuration Module Programmer Kit (Part Number 050-03311-0002) in conjunction with a personal computer. See the configuration module user data for detailed setup instructions. Follow the instructions for the programmer.

Perform radar R/T calibration as follows: Reference AlliedSignal (Bendix/King) 2000 Radar Sensor installation manual.

a) Antenna Clearance Check

1] From the Maintenance Set Up Page Select RADAR.

2] Set the radar park position to Full Up. Enter SAVE.

3] Restart the MFD. (SYSTEM) (R5)

4] From the Radar Page, set the Radar Function to SBY. (L3)

5] Set the Radar Mode to GND. (R1) Set KNOB to gain (L5)

6] Reduce the gain until the gain indicator shows the minimum setting.

7] Set the Antenna Tilt to full Up (U 15.0)

8] Set Range to 240 NM.

9] From the Aux Page, enter Maintenance Mode, as described in Section 6.3, “Using the Maintenance Mode Page” on page 25.

10] Press “Radar Setup”.

11] Press “Calibration” to display the RT CALIBRATION DATA page. Upon display of the RT CALIBRATION DATA page with the system in calibration mode, all fault fields will flash briefly. This verifies that the system is in calibration mode.

12] Starting at -30, slowly adjust the gain control downward until the antenna clearance scan begins. This should take place at a gain value no lower that -26. The antenna will move to each of the extreme positions to determine that there is no interference with antenna movement and all scan motors are working properly.

b) Calibrate Radar Pitch and Roll.

Use the right outer knob to set ROLL TRIM to 0 degrees.

Calibrate AHRS ARINC 429 Pitch Offset.

1] Level aircraft.

2] Adjust the GAIN buttons for a GAIN POT /2 setting to 11- or 12-.

3] Set the PITCH ANGLE field to 0.0 +/- 1.0 degrees.

To increment the value of the PITCH ANGLE, use the left inner knob to change the TILT SETTING to between 5 and UP. When the TILT SETTING is set to between 5 and 10 UP, the value of the PITCH ANGLE field will slowly increase.

To decrement the value of the PITCH ANGLE, use the left inner knob to change the TILT SETTING to between 5 and 10 DOWN. When the TILT SETTING is set to between 5 and 10 DOWN, the value of the PITCH ANGLE field will slowly decrease.

When the desired setting is reached, quickly adjust the TILT SETTING to above 10 UP to lock in the setting.

c) Calibrate AHRS ARINC 429 Roll Offset.

1] Verify that the aircraft is level.

2] Adjust the GAIN buttons for a GAIN POT setting to 7- or 9-.

3] Set the ROLL ANGLE field to 0.0 +/- 1.0 degrees.

To increment the value of the ROLL ANGLE, use the left inner knob to change the TILT SETTING to between 5 and 10 UP. When the TILT SETTING is set to between 5 and 10 UP, the value of the PITCH ANGLE field will slowly increase.

To decrement the value of the ROLL ANGLE, use the left inner knob to change the TILT SETTING to between 5 and 10 DOWN. When the TILT SETTING is set to between 5 and 10 DOWN, the value of the ROLL ANGLE field will slowly decrease.

When the desired setting is reached, quickly adjust the TILT SETTING to above 10 UP to lock in the setting.

d) Save Configuration:

1] Adjust the GAIN controls for a GAIN POT /2 setting to 4- or 5-.

2] The FAULTS field will display GYRO.

3] Set the TILT SETTING to 15.0 DOWN. The fault fields will flash indicating that your settings are being saved. If the save procedure is successful the GYRO fault will disappear and the azimuth count will cycle through its entire number range.

NOTE: Transmitter FAULTS with the Radar OFF.

4] If the GYRO fault remains, set the TILT to 0 and repeat the previous step.

5] Enter EXIT.

6] After the Calibration is completed then enter SAVE.

7] Return to the RADAR Setup Page and change Park Position to Centered and SAVE.

(h) Replace 7.3.8, Map Heading, with the following:

1) From the maintenance page, depress LSK (R1) to access the MAP Heading page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration; configure the Map Heading page for the following conditions:

Map Heading:	FMS/GPS

3) Press SAVE button.

(i) Replace 7.3.10, Engine Sensor, with the following:

1) From the maintenance page, depress LSK (L4) to access the Engine Set-up page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration. Configure the Engine Setup page for the following conditions:

Aircraft Model:	PA-46-350P Series
Serial Port:	RS232 4
CK:	☑ Vacuum Sys. Inst. (PA-46R-350T only with deice)

3) When setting up a MFD for the first time only the following set up box appears. “Select ACFT Model and Port (s)” Press “Resync”.

(j) Under 7.3.10.1, Engine Sensor Setup, replace “Fuel Quantity Calibration (Optional)” with Fuel Quantity Indicator Calibration (S/N’s 4636375 and up.) (S/N’s 4692001 and up.), in 28-40-00

(k) Replace 7.3.11.1, Auxiliary Data; 7.3.12, Datalink; 7.3.12.1, Narrowcast Datalink Setup; and 7.3.12.2, Broadcast Datalink Setup with the following:

1) From the maintenance page, depress LSK (R2) to access the AIRCRAFT Set-up page.

2) Using the right outer concentric knob to select the configuration field and the right inner concentric knob to select the desired configuration. Configure the Aircraft Setup page for the following conditions:

Narrowcast:	Quake SC or Not Installed
Port:	RS232 6 (Data link Default)
Broadcast:	XM Radio (PA-46-350P) / Sirius Radio (PA-46R-350T)
Port:	RS232 1
Aux Data:	Entegra PFD
Port:	ARINC 429 RX 4
Dimming Bus Voltage:	Shows Current Voltage with aircraft Day/Night switch in “Night”
Brightest Dimming Voltage:	24.0 Volts
Darkest Dimming Voltage:	5.0 Volts

3) Press SAVE button.

(l) See "Chart 1" for Port Information.

CHART 1
PORT INFORMATION PAGE

Port	Used By PA-46-350P / PA-46R-350T	Port	Used By PA-46-350P / PA-46R-350T
RS232 1	Broadcast	ARINC 429 RX3	Traffic / Blank
RS232 2	Blank / Traffic	ARINC 429 RX4	PFD Link
RS232 3	Lightening / TWX-670	ARINC 429 TX1	Radar / Blank
RS232 4	Engine	ARINC 429 TX2	TAWS / Blank
RS232 5	Datalink	ARINC 429 TX3	Blank
RS232 6	Datalink	ARINC 429 TX4	Blank
ARINC 429 RX1	GPS 1	ARINC 453-1	Radar / Blank
ARINC 429 RX2	GPS 2	ARINC 453-2	TAWS / Blank

(3) Data Acquisition Unit (DAU)

The DAU is mounted underneath the instrument panel on the right aft side of the forward pressure bulkhead (F.S. 100.00) (see "Figure 1" on page 342210). Maintenance is on condition.

NOTE: Whenever the DAU is replaced, perform Fuel Quantity Indicator Calibration (Avidyne Entegra) in 28-40-00.

(a) Use 200-00041-000 DAU ICA, Avidyne Document No. AVSIU-011, with the following exceptions:

1) In para 6, in “Table 2 - DAU Pinout,” pins J1-2 and J1-21 have “No Connection” in the Piper installation.

2) In para 6, in “Table 4 - DAU Sensor Compatibility,” parameter “VAC” is not used in the Piper installation.

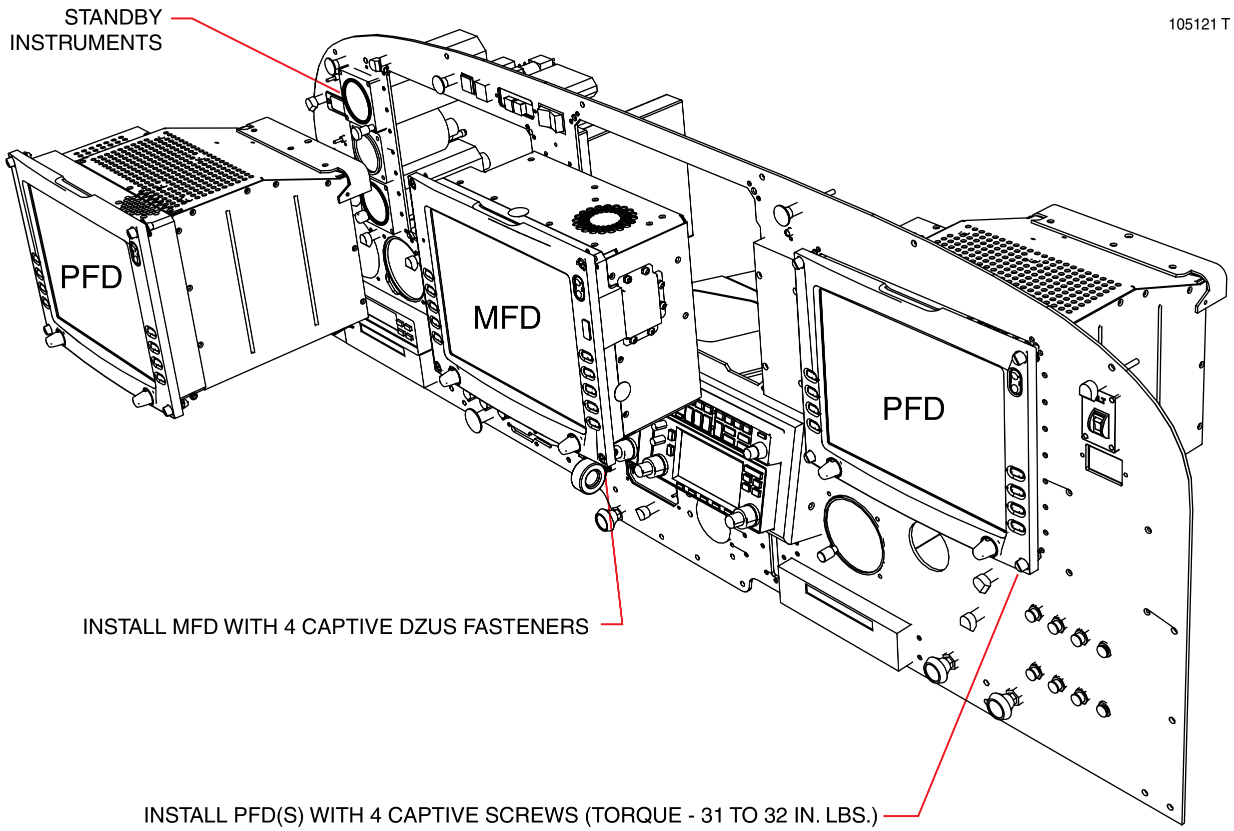
(b) Configuration/calibration procedures are in 28-40-00 under Fuel Quantity Indicator Calibration (Avidyne Entegra).

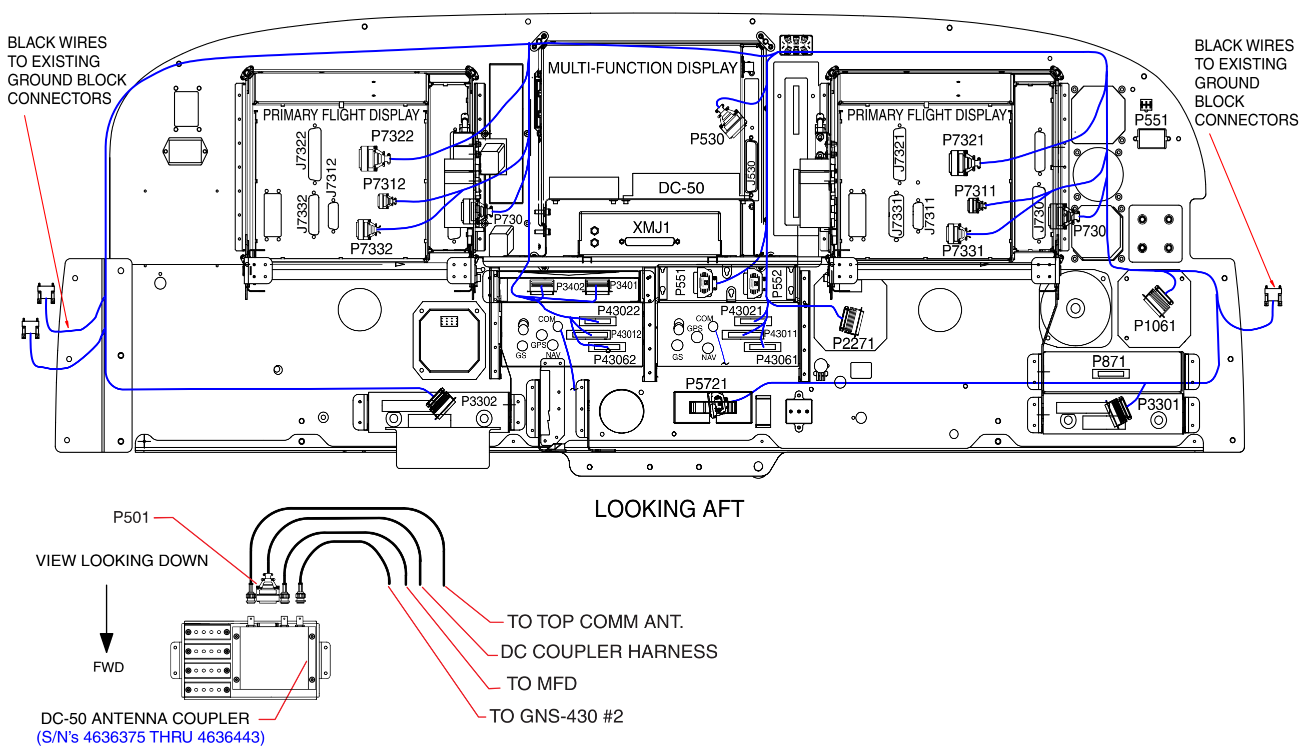
(4) Magnetometer/OAT Sensor Assembly

See information under Primary Flight Display, above. Removal, installation, and calibration instructions are provided under "Magnetic Heading Systems" on page 342213.

C. Component Locator

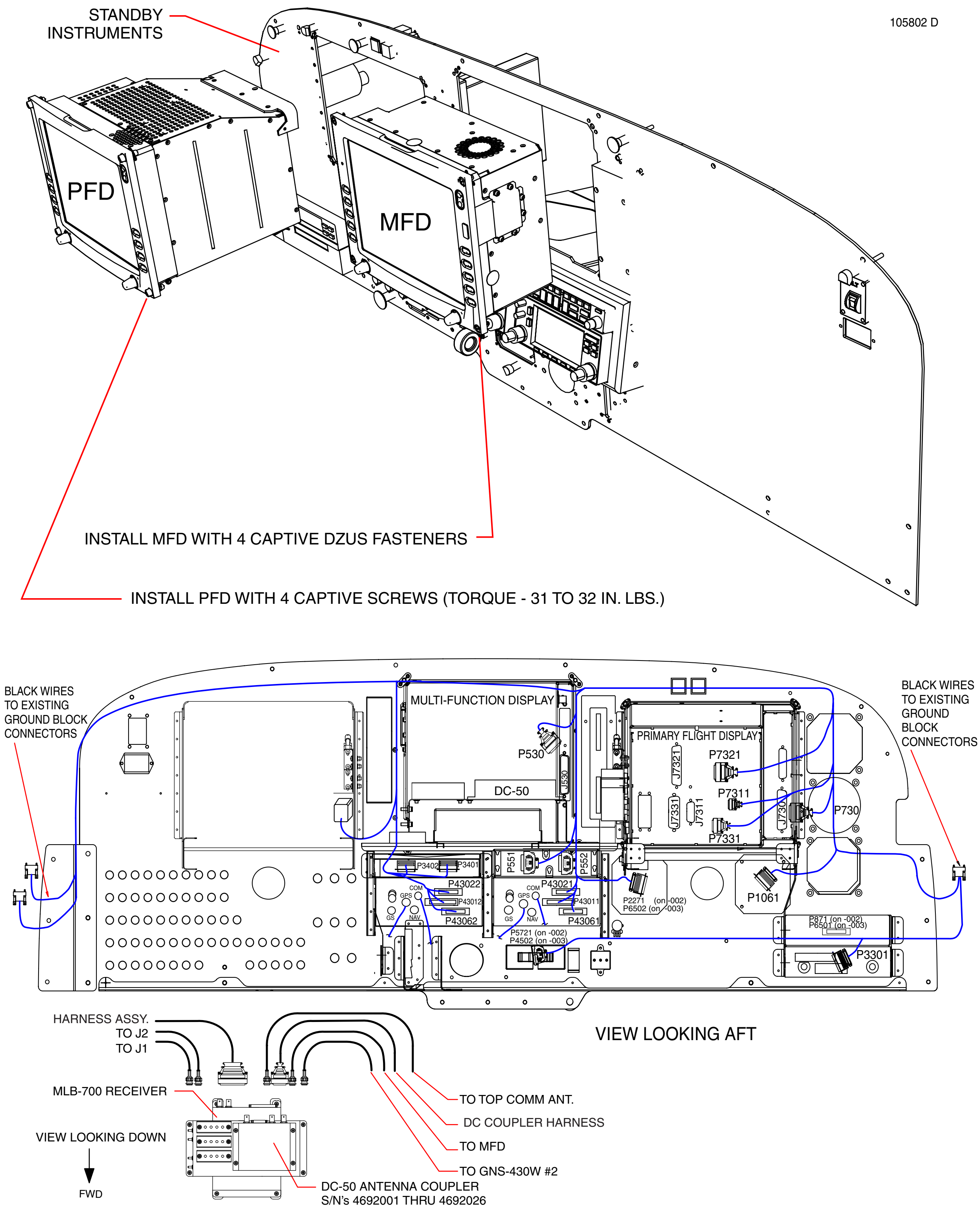
See "Figure 1" on page 342210 and "Figure 2" on page 342214.





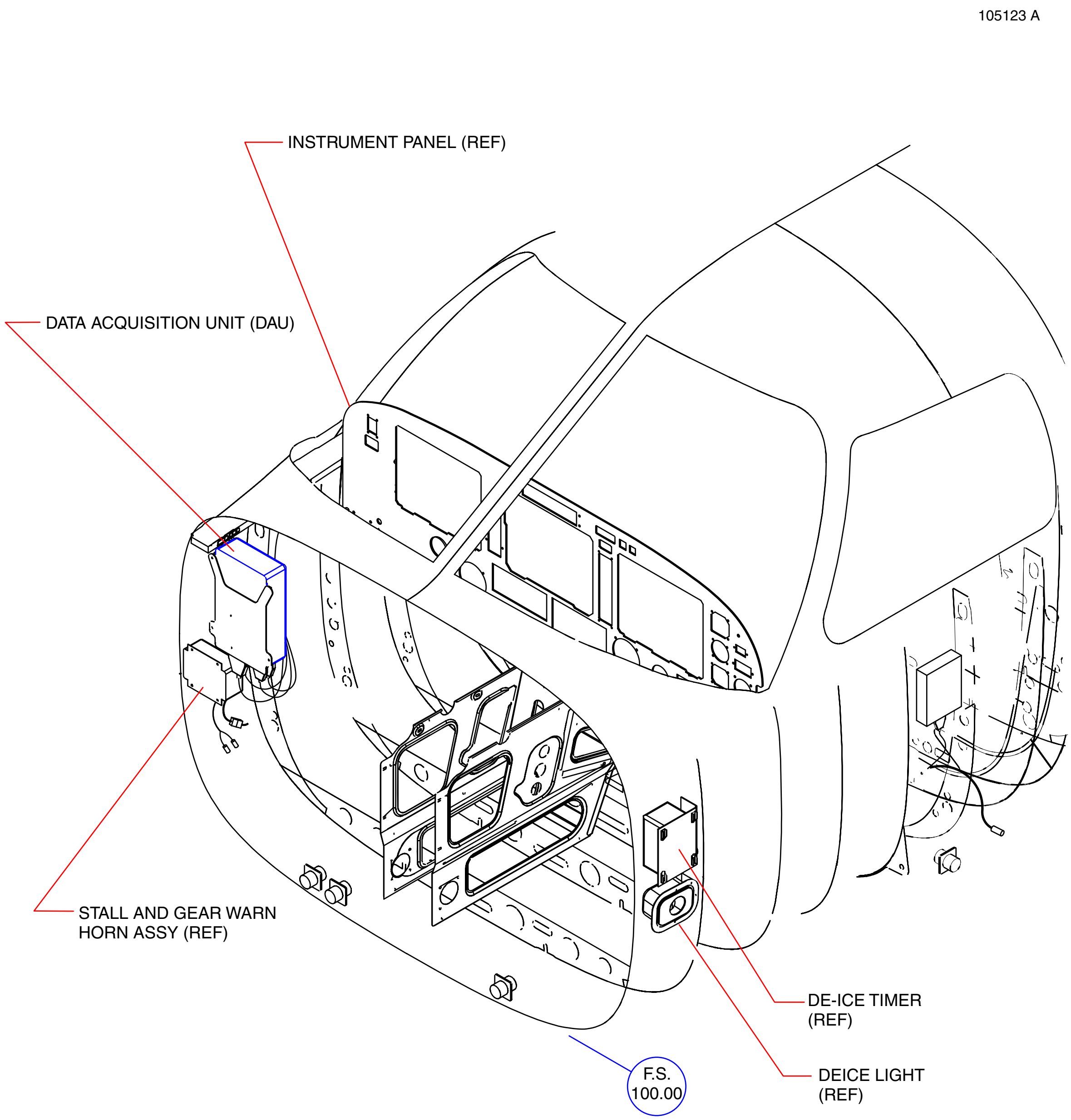
Effectivity
PA-46-350P

Avidyne Entegra Component Locator
Figure 1 (Sheet 1 of 3)



Avidyne Entegra Component Locator
Figure 1 (Sheet 2 of 3)

Effectivity
PA-46R-350T



Avidyne Entegra Component Locator (Typical)
Figure 1 (Sheet 3 of 3)

D. Magnetic Heading Systems

The Magnetometer / OAT Sensor Assemblies (Mag/OAT) feed magnetic heading information to the Primary Flight Display (PFD) and the Multifunction Display (MFD). The Mag/OAT's are mounted under the dorsal fairing atop the aft fuselage at F.S. 282.372. See "Figure 2" on page 342214.

NOTE: The OAT sensors in the Mag/OAT assemblies are not used. See "Outside Air Temperature" on page 341026.

(1) Magnetometer/OAT (MAG/OAT) Assemblies

(a) Removal

1) Remove the dorsal fairing to expose the magnetometers.
2) Disconnect the wiring harness(es) from the aft fuselage harness.
3) Unscrew the three screws securing the OAT probe mounting plate to the mounting bracket.
4) Remove the OAT probe mounting plate and unscrew and remove the OAT probe nut(s) from the bottom of the mounting plate and remove the OAT probe(s).
5) Unscrew and remove the three brass screws and washers securing the magnetometer to its mounting bracket and remove the magnetometer(s).

(b) Installation

CAUTION: THE MAGNETOMETERS ARE SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Place the magnetometer(s) into position on its mounting bracket, with the arrow on the magnetometer pointing forward, and secure with brass screws and washers (3 ea).
2) Place the OAT probe(s) into position on its mounting plate and secure with nut(s) from bottom.
3) Place OAT probe mounting plate into position on mounting bracket and secure with brass screws and washers (3 ea.)
4) Connect wiring harness(es) to the aft fuselage harness.
5) Reinstall the dorsal fairing.

NOTE: Ensure correct hardware is used when reinstalling vertical fin dorsal fairing over flux detector.

(2) Calibration

(PIR-PPS60227, Rev. B. / PPS60227-1, Rev. New. / PPS60232, Rev. B.)

Calibration can be accomplished using either of the following two methods.

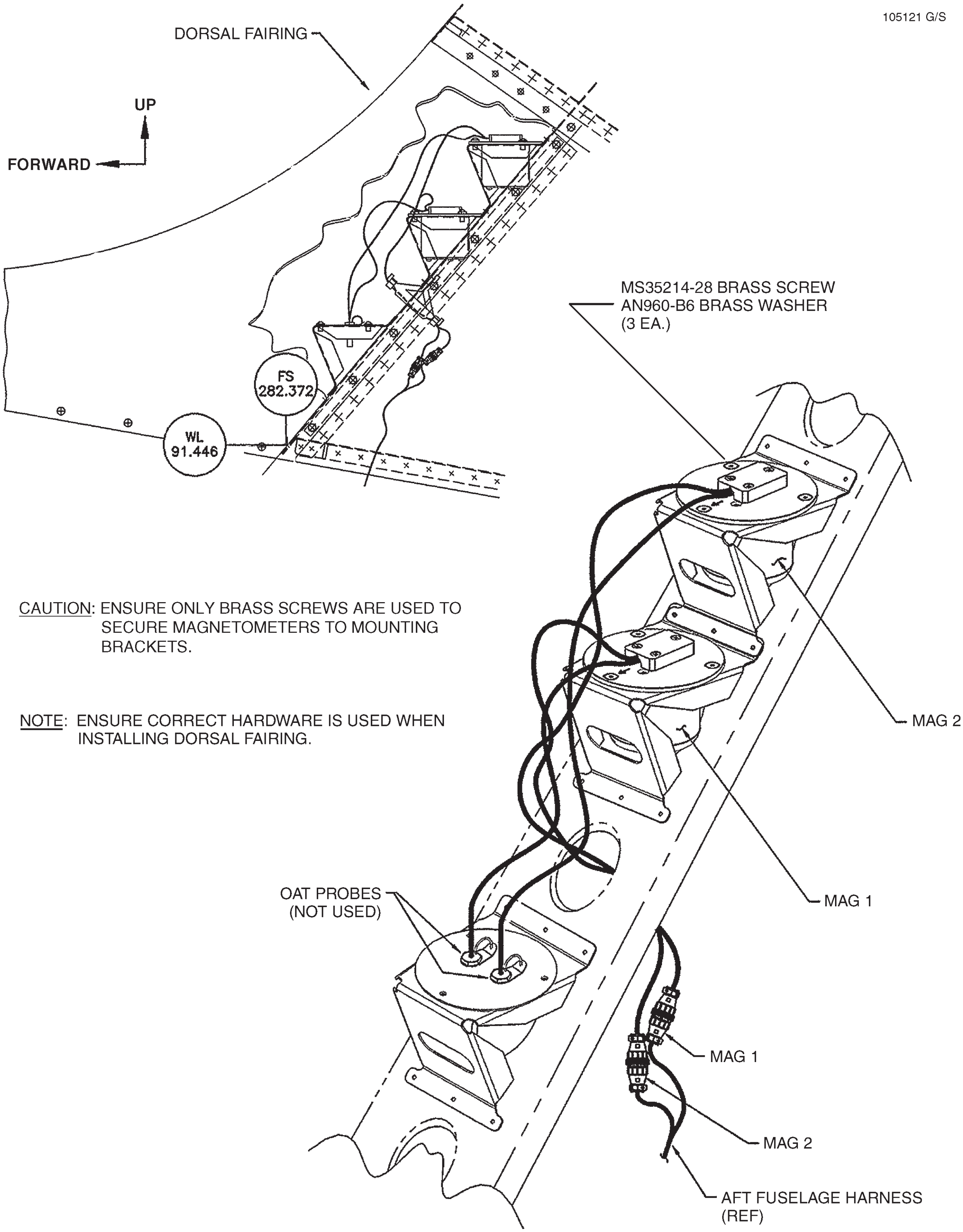
NOTE: LSK L1 "Perform IRU Mx" and LSK L3 "Perform ADU Cal" are to be performed by Avidyne Service Technicians only. These functions are for bench test only.

(a) Bore sight reference compass (requires two (2) technicians)

NOTE: A hand-held bore sight reference compass is required for this procedure.

1) Apply external power to the aircraft to allow the PFD(s) to align while conserving the battery for the remaining calibration procedure.
2) Allow three (3) minutes for the PFD(s) to align, until the ADAHRS countdown timer expires and the Gyro Warm-up box is removed from view.

NOTE: The PFD countdown timer might automatically restart its count before the countdown expires. This is a normal operation and does not indicate a problem with the PFD.

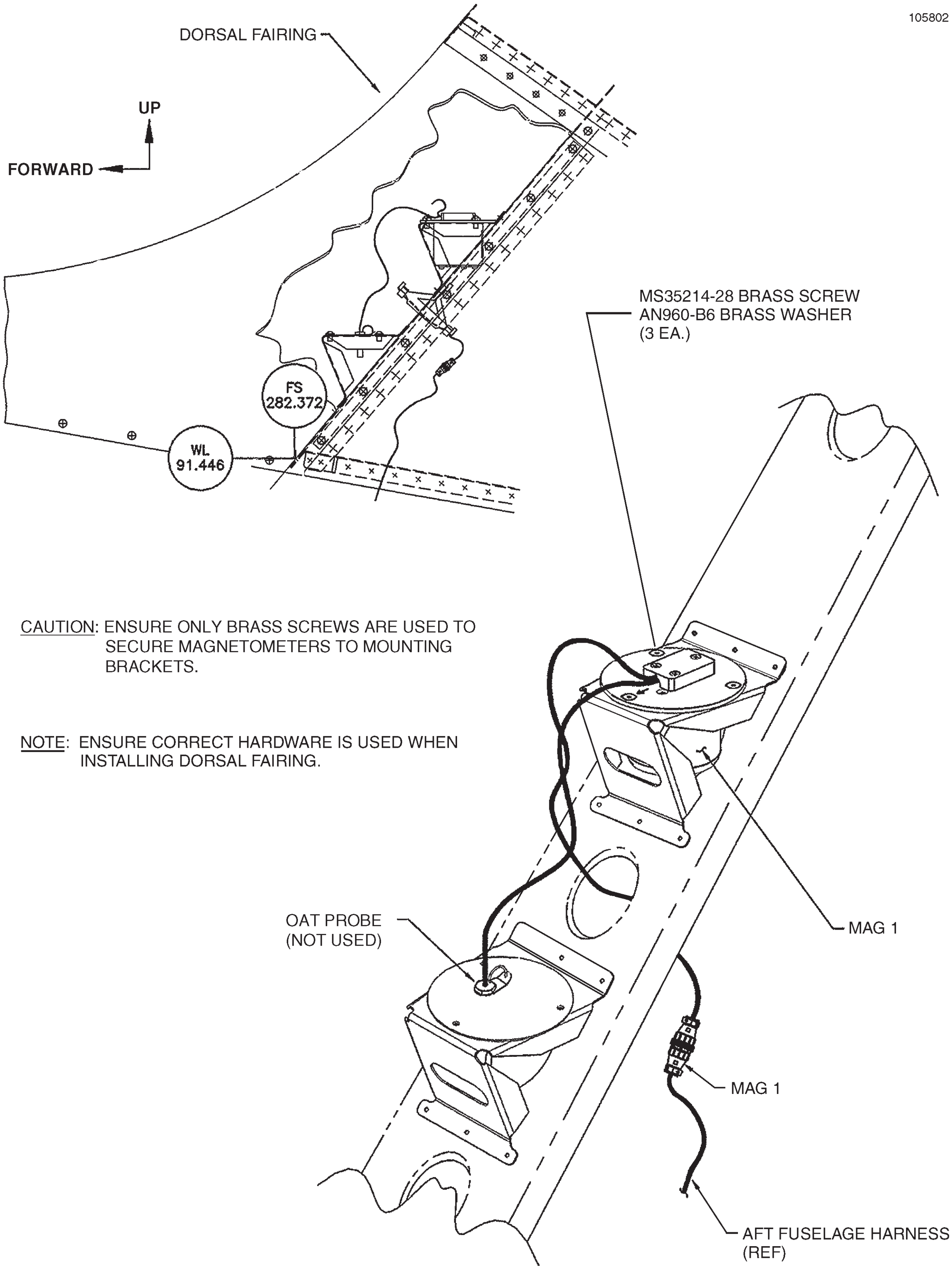


CAUTION: ENSURE ONLY BRASS SCREWS ARE USED TO SECURE MAGNETOMETERS TO MOUNTING BRACKETS.

NOTE: ENSURE CORRECT HARDWARE IS USED WHEN INSTALLING DORSAL FAIRING.

Effectivity
PA-46-350P

Magnetometer/OAT Sensor Assemblies
Figure 2 (Sheet 1 of 2)



CAUTION: ENSURE ONLY BRASS SCREWS ARE USED TO SECURE MAGNETOMETERS TO MOUNTING BRACKETS.

NOTE: ENSURE CORRECT HARDWARE IS USED WHEN INSTALLING DORSAL FAIRING.

Magnetometer/OAT Sensor Assemblies
Figure 2 (Sheet 2 of 2)

Effectivity
PA-46R-350T

3) Allow an additional 10 minutes in the aligned state to ensure complete stability.
4) Place Battery Master Switch “ON”.
5) Remove external power from the aircraft.
6) Position the aircraft in an area located away from metal structures and magnetic interference such as that caused by underground power lines, etc.
7) Using the bore sight compass, align the aircraft to a heading of 360° +/- 1°.
8) Enter the System Setup mode on the PFD(s).
9) Press the “Perform Mag Cal” Line Select Key (L2).
10) Follow the directions on the “Magnetometer Calibration Page” and press the “Calibrate Heading” button and wait until “Done” is displayed.
11) Align the aircraft to each of the remaining 11 consecutive 30° headings using steps (g) through (j) and press the “Calibrate Heading” LSK (R4) at each of these alignment points after satisfactory alignment is achieved.
12) At the final alignment point (330°), remain stationary until “Done” is displayed. This could take up to one minute as data from the 12 calibration points is loaded onto the PFDs.
13) When complete, the following message appears:

MAG CAL COMPLETE - YOU MUST CYCLE POWER ON THE PFD IN ORDER FOR THE MAGNETOMETER CALIBRATION TO TAKE EFFECT.

14) Cycle power by pulling the following circuit breakers.

		Dual PFD Installation	**Single PFD Installation**
a)	Main bus:	Pilot PFD and Copilot PFD	PFD
b)	Non Essential Bus:	PFD/MFD Power Converter	
c)	Avionics 1 Bus:	Pilot PFD	
d)	Avionics 2 Bus:	Copilot PFD	

NOTE: Wait 20 seconds before applying power.

15) Verify the calibration by aligning the aircraft with consecutive 90° headings, starting with 360°. Use the same alignment procedure as before and view the digital heading display on the PFD(s) at each of these headings.

(b) Magnetometer Calibration Using Compass and Engine Run

1) Perform normal engine start and turn on Avionics Master if needed for taxi communications.
2) Remain stationary and allow the PFD(s) to align for approximately 3 minutes until the ADAHRS countdown timer expires and the Gyro Warm-up box is removed from the PFD screen.

NOTE: The PFD countdown timer might automatically restart its count before the countdown expires. This is a normal operation and does not indicate a problem with the PFD.

3) Allow an additional 10 minutes in the aligned state to ensure complete stability.
4) Taxi to the compass rose with a technician to act as a spotter to aid in the alignment.
5) Enter the System Setup mode on the PFD(s) .
6) Press the “Perform Mag Cal” Line Select Key (L2).
7) Align the aircraft with magnetic north on the compass rose with the assistant to within +/- 1°.

8) Follow the directions on the “Magnetometer Calibration Page” and press the “Calibrate Heading” button and wait until “Done” is displayed.

9) Align the aircraft to each of the remaining 11 consecutive 30° headings using steps "7)" through "8)" and press the “Calibrate Heading” LSK (R4) at each of these alignment points after satisfactory alignment is achieved.

10) At the final alignment point (330°), remain stationary until “Done” is displayed. This could take up to one minute as data from the 12 calibration points is loaded onto the PFD(s) .

11) When complete, the following message appears:

    MAG CAL COMPLETE - YOU MUST CYCLE POWER ON THE PFD IN ORDER FOR THE MAGNETOMETER CALIBRATION TO TAKE EFFECT.

12) Cycle power by pulling the following circuit breakers.

		Dual PFD Installation	Single PFD Installation
a)	Main bus:	Pilot PFD and Copilot PFD	PFD
b)	Non Essential Bus:	PFD/MFD Power Converter	
c)	Avionics 1 Bus:	Pilot PFD	
d)	Avionics 2 Bus:	Copilot PFD	

NOTE: Wait 20 seconds before applying power.

13) Verify the calibration by aligning the aircraft with consecutive 90° headings, starting with 360°. Use the same alignment procedure as before and view the digital heading display on the pilot and copilot PFDs at each of these headings.

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ASPEN AVIONICS EVOLUTION BACKUP DISPLAY (EBD)

Aspen Avionics Evolution Backup Display (EBD)

Installed as standard equipment in S/N's 4636633, 4636652–4636775.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. Description

The Aspen Avionics EBD system is an electronic backup display for the primary EFIS system. The EBD contains an internal ADAHRS that is used to provide attitude and heading data for the display. The EBD as installed by Piper is connected to the emergency power buss and is powered by the emergencey battery to provide a minimum of 30 minute operation in the event of power loss under all foreseeable operating conditions. In addition, the EBD contains an internal battery which will provide an additional nominal 30 minute operation in the event emergency battery exhaustion.

Additional equipment is normally installed in support of the displays, including the Remote Sensor Module (RSM) and Configuration Module (CM).

The EBD system is configured as a basic system in which the EBD provides display of attitude, airspeed, altitude, direction of flight, vertical speed, turn rate, and turn quality. The system provides display of pilot-selectable indices ("bugs"), and annunciations to increase situational awareness and enhance flight safety.

NOTE: The terms EFD1000, EFD and EBD are used interchangeably.

B. Internal Battery Operation

When IAS is greater than 30 KIAS and the input voltage drops below 25.6 volts the EBD will automatically switch to internal battery (e.g., it assumes aircraft charging system failure).

Whenever indicated airspeed is invalid or below 30 KIAS (i.e., on the ground) the EBD will power up and power down with the application or removal of external power. A message is presented during the normal power down sequence to enable the pilot to abort the shutdown and switch to internal battery.

C. Troubleshooting

See "Chart 1" on page 34242.

D. Inspections

(1) Annual

All units, brackets, installation hardware and wiring of the EBD system should be checked as defined below during annual inspection. Items found to be defective should be repaired or replaced prior to returning the aircraft to service.

(a) EBD Inspection

Inspect the EBD for damage and operation. Visually inspect the EBD wiring for damage, chafing, or excessive wear. The EBD braided bonding strap should be checked for proper termination at the EBD and aircraft grounding point to maintain HIRF and Lightning compliance.

Verify $\leq$ 3 milliohms from EBD ground stud to airframe ground. The installation of the EBD should be inspected for corrosion on the EBD and the structure it is mounted on. The fasteners should be inspected for tightness and general condition.

CHART 1 (Sheet 1 of 5)
TROUBLESHOOTING - EBD

Initialization Page Tests

Test	Cause	Remedy
IOP.	a) Fail. b) System reboots after IOP test.	a) Replace EBD. b) Replace EBD.
Config Module (CM).	a) Fail. b) Wrong CM version. c) System reboots after CM Test. d) Displays “Initializing” for more than 20 seconds.	a) Check CM wiring. Replace CM. b) Install correct SW version CM. c) v2.0 or v2.1 display installed with a v2.2 CM. Install correct CM or EBD. d) CM unplugged or miswired.
RSM.	Fail (x).	Check RSM to EBD wiring for shorts or opens. Repair or replace RSM. Repair or replace EBD.
IMU.	Fail.	Replace EBD.
ADC.	Fail.	Replace EBD.
ADAHRS.	a) Fail. b) “Initializing” for more than 3 minutes. c) “Initializing” for more than 3 minutes with a RSM Fail above.	a) Replace EBD. b) Remove Pitot and Static line from back of EBD and reboot. If problem still exists then replace the EBD. If problem clears then repair Pitot or Static obstruction/kink. c) Repair RSM wiring or replace RSM.

NOTE: With software version 2.3 and later it is possible to advance the system beyond a failed Initialization Screen by holding the MENU key and line select keys #1 and #2 simultaneously for 3–4 seconds. The Diagnostic screen may then be accessed in the installation menu for further troubleshooting.

CHART 1 (Sheet 2 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting

Trouble	Cause	Remedy
Display does not power on. (Note: there can be up to a 20 second delay from the application of power to a visible display.)	a) EBD missing A/C power.	a) Check EBD circuit breaker, EBD on/off switch on panel, wiring, and A/C battery voltage > 11.5 volts.
	b) EBD may have been improperly shut down.	b) Switch unit off using “REV” button or “SHUT DOWN” command from Main Menu page 6.
	c) EBD missing A/C ground.	c) Check wiring to EBD.
	d) EBD is defective.	d) Repair or replace EBD.
Display does not power off. (Note: EBD will switch to internal battery if airspeed is greater than 30kts.)	a) Airspeed is above 30kts.	a) Normal operation.
	b) EBD may have been switched to internal battery.	b) Switch unit off using “REV” button or “SHUT DOWN” command from Main Menu page 6.
	c) EBD may have been improperly shut down.	c) Hold “REV” button for 20 seconds or unplug EBD internal battery for 3 seconds.
	d) EBD is defective	d) Repair or replace EBD.
“CONFIG MODULE LINK FAIL” message.	a) Configuration Module unplugged or miss wired.	a) Check CM plug and wiring from EBD to CM.
	b) Configuration module defective.	b) Repair or replace CM.
	c) EBD defective.	c) Repair or replace EBD.
“RSM LINK FAIL” message.	a) RSM to EBD communication lost.	a) Check RSM to EBD wiring for shorts or opens.
	b) RSM failed.	b) Repair or replace RSM.
	c) EBD failed.	c) Repair or replace EBD.
“WRONG CONFIG MODULE” message.	EBD is at one software level and config module is at a different software level.	Convert config module per appropriate service bulletin.

CHART 1 (Sheet 3 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued

Trouble	Cause	Remedy
ALTIMETER, AIRSPEED, VSI FAIL (RED-X).	a) Air data sensor has not had sufficient warm-up time.	a) Allow up to 20 minutes at temps below -20°C for flags to clear.
	b) Pitot/static lines reversed.	b) Connect pitot line to “P” port and static line to “S” port on EBD.
	c) Air data sensor failed.	c) Repair or replace EBD.
Vmo Fail.	Vmo configured incorrectly.	Re-check MOL programming.
ATTITUDE FAIL or DIRECTION FAIL (RED-X). (Note: Attitude flags could take up to 3 minutes to clear at temps below -20 °C).	a) AHRS sensor has not completed initialization.	a) Allow up to 3 minutes for AHRS to initialize.
	b) RSM failed/data missing.	b) Check RSM to EBD wiring. Repair or replace RSM.
	c) Pitot and/or Static lines crossed, unplugged, or blocked.	c) Correct pitot/static plumbing. issue.
	d) EBD is defective.	d) Repair or replace EBD.
ATTITUDE FAIL and DIRECTION FAIL associated with “CHECK PITOT HEAT” message.	In Flight, Normal if pitot blockage due to ice or other.	Use pitot heat or check pitot system for blockage.
CROSS CHECK ATTITUDE message (yellow) (also see sluggish or poor AHRS (ADI) performance below)	a) If it occurred on system start.	a) RESET AHRS.
	b) Normal after abrupt maneuvers on ground or in air.	b) RESET AHRS.

CHART 1 (Sheet 4 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued

Trouble	Cause	Remedy
"ERRONEOUS CALIBRATION VALUES" message during RSM Cal or Excessive Heading errors in one quadrant, or errors that are higher than actual in some quadrants and lower than actual in other quadrants.	a) RSM is tilted more than: ±4° to the aircraft longitudinal axis; ±10° to the lateral (wings level) axis; ±10° to the zero pitch axis.	a) Shim RSM to within limits stated.
	b) Poor RSM calibration.	b) Re-run RSM calibration at constant rate turns on flat ground.
	c) RSM calibrated too close to buildings or ferrous objects.	c) Re-run RSM calibration away from buildings and other ferrous objects.
	d) Ferrous hardware used to mount RSM.	d) Only non-ferrous screws, nuts, washers may be used on RSM.
	e) Airframe or external magnetic interference.	e) Check for magnetized areas on airframe close to RSM. Verify no ferrous hardware is near RSM. Degauss magnetized area(s).

CHART 1 (Sheet 5 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued

Trouble	Cause	Remedy
Sluggish or Poor AHRS (ADI) performance. Poor AHRS performance in steep bank turns. Sluggish compass card. (Note: May or may not be associated with "Cross Check Attitude" message.	a) RSM magnetic interference.	a) Survey RSM location using handheld compass. Move the compass around the location looking for needle deflection. There should be no more than 2° of compass needle movement within an area 18" x 18" around the location. If excessive needle movement is found, degauss the affected area.
	b) RSM has become magnetized.	b) With power removed from EBD system degauss RSM and general area.
	c) "Pitch Attitude Trim" or "Panel Tilt Pitch Compensation" adjustment made without performing a subsequent RSM Calibration.	c) Perform an RSM Calibration.
	d) Pitot and/or Static line connections at EBD blocked, kinked, or unplugged.	d) Check pitot/static connections and plumbing for blockage. Check IAS and ALT sensor per Ground Test Procedure.
	e) Normal after abrupt maneuvers.	e) Perform AHRS Reset.
Excessive Heading Lead / Lag during or after turns (>7°).	Magnetic Interference.	Check for RSM magnetic interference as in "Sluggish or Poor AHRS (ADI) performance", above. If that does not correct the issue, contact Piper Customer Service.

(b) EBD Internal Battery Test

Perform the EBD Internal Battery Test under Ground Test Procedure.

(c) RSM Inspection

Visually inspect the RSM for damage and wear on the bonding strap. RSM wiring should be checked for damage, chafing, or excessive wear. Verify RSM bonding from the ground stud to access plate ground is ≤ 3 milliohms to maintain HIRF and Lightning compliance. The RSM installation should be inspected for corrosion on the RSM and the wing access plate. The installation should be inspected for cracks and loose or damaged fasteners.

(d) Configuration Module Inspection

Visually inspect the Configuration Module(s) for damage. Visually inspect the Configuration Module wiring for damage, chafing, or excessive wear.

(2) Each Twenty-Four (24) Calendar Months

Use procedures in Aspen Avionics Tech Note 2009-09, Rev. B or later, Document Number 991-00023-001, Tech Note for Air Data Calibration to test and inspect the static pressure system and altimeters. Ensure compliance with the requirements of FAR 43, Appendix E. (See FAR 91.411.)

(3) Each Three Years or 2,200 Hours, Whichever Comes First

Replace the EBD internal battery per EBD Battery Replacement under Components, EBD Display, below.

E. Components

(1) EBD Display

The EBD is a digital system that consists of a high resolution 6-inch diagonal color LCD display, user controls, photocell and Micro SD data card slot. The rear portion of the EBD includes a non-removable electronics module which contains a full air data computer, attitude heading reference system, power supplies, backup battery, and dual processor electronics. Also on the rear of the unit, a fan is provided to cool the backlight and electronics.

The EBD mounts to the front surface of the instrument panel. The electronics module and cooling fins on the back are sized to fit into instrument panel holes. A recess-mount bracket mounts the displays nearly flush with the instrument panel.

The instrument is installed in a vertically oriented pair of instrument openings, without interfering with the surrounding instruments. The EBD contains a microSD card port and reader at the bottom of the display bezel. When authorized, software updates and system upgrades can be installed using the card port. The port is also used for database information in the EBD MFD.

The EBD is a digital system and supports the ARINC 429 digital interface. The PFD connects directly to the interfaced equipment.

(a) Removal

Verify power is off. Carefully insert a flat blade screw driver into the locking mechanism on the top center of the EBD. While gently prying pull back the top of the EBD and extract from bracket. Remove nut securing braided ground strap to EBD. Remove pitot and static quick connectors (EBD only) by pulling back outer spring loaded locking sleeve while unplugging connectors. To remove 44 pin D-sub connector unscrew both jackscrews fully and pull connector straight back.

(b) Installation

Verify power is off. Install 44 pin D-sub connector and tighten jackscrews until connector is fully seated. Install pitot and static lines (EBD only) to back of EBD by firmly pressing the fitting until fully seated (pitot and static quick connectors are keyed and cannot be crossed). Gently pull on connector to ensure proper connection. Connect braided bonding strap to EBD with nut. Insert bottom of EBD into bracket and pivot top forward until it locks into place on bracket.

EBD Standby Display
Figure 1

(c) Return to Service

1) Verify all system interfaces are functional.

2) Verify proper bonding per "Bonding Check" on page 342413.

3) Perform the "System Leak Test" on page 342431.

(d) EBD Battery Replacement

EBD battery replacement must only be performed by a properly certified individual or facility. Remove EBD from panel as above. Remove two screws on each end of the football shaped cover plate on rear of the EBD. Unplug electrical connector and slide battery out of EBD. Install new battery in EBD then connect battery plug. Replace cover plate and tighten the cover screws. Tighten to 12 in-lbs. Reinstall and test EBD as above.

(e) EBD Knob Replacement Procedure

The knob(s) are removed by gripping the knob with the fingers of your hand and then pulling straight off the shaft using moderate force. Install the new knob P/N 520-00020-001 by pushing the knob on to the shaft with the fingers of your hand until it bottoms out. After knob replacement power up the system and verify the knob can turn freely and that the “push to enter” function operates properly by accessing the HDG bug or CRS pointer function.

(2) Remote Sensor Module (RSM)

The RSM is required for each EBD and connects directly to the display. It physically resembles a traditional GPS antenna.

The RSM contains all of the sensors that must be remotely located from the EBD display unit. It is mounted in a magnetically quiet environment. The RSM is installed internally on an access cover on the underside of the left wing.

The RSM is powered by the EBD through a shielded wire harness and contains the Magnetic “flux” sensors.

(a) Removal

CAUTION: DO NOT USE A MAGNETIC TIPPED SCREW DRIVER WHEN REMOVING AND REPLACING THE RSM.

1) Verify power is off.
2) Remove screws (12 ea.) securing access cover to wing skin. Take care to support access cover as it becomes detached from wing.
3) Disconnect RSM electrical connector from wing wiring harness and remove RSM and access cover from airplane.
4) To remove RSM from mounting bracket, first disconnect ground wire from mounting bracket stud. Then remove four (4) non-ferrous screws, washers, and nuts from RSM and separate RSM from mounting bracket taking care to feed wiring pigtail through gap in bracket.

(b) Installation

CAUTION: DO NOT USE A MAGNETIC TIPPED SCREW DRIVER WHEN REMOVING AND REPLACING THE RSM.

1) Verify power is off.
2) If RSM has been separated from its mounting bracket, feed wiring pigtail through gap in bracket and place RSM in bracket with flat end forward (see arrow on mounting bracket). Secure with four (4) non-ferrous screws, washers, and nuts. Torque as specified in "Figure 2".
3) Secure the RSM ground wire to stud on mounting bracket.
4) Connect RSM electrical connector to wing wiring harness.
5) Verify proper bonding per Bonding Check under Post Installation Setup Procedure, below.
6) Place RSM and access cover in position at wing skin opening.
7) Secure access cover to wing skin with non-ferrous screws (12 ea.)

(c) Return to Service

Perform "RSM Calibration" on page 342424.

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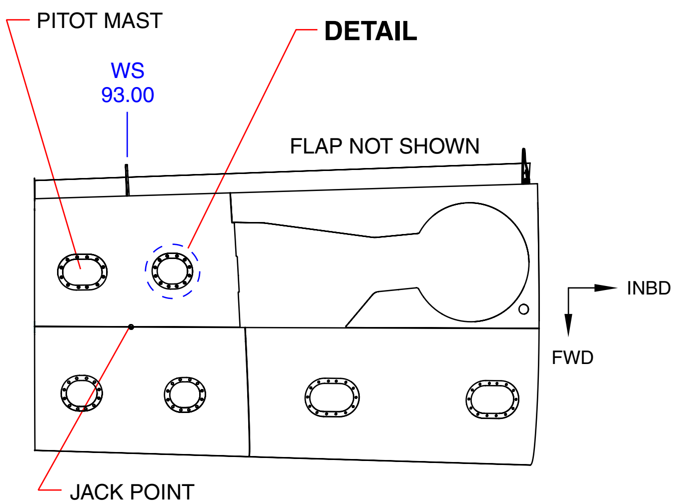
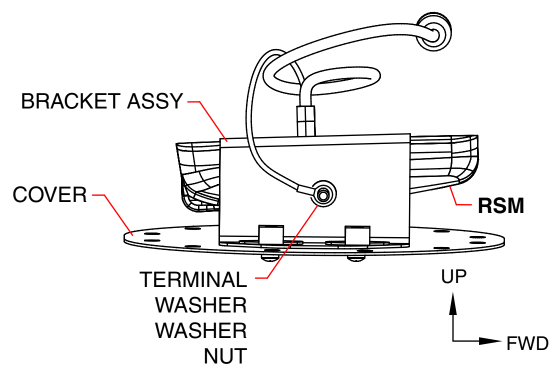
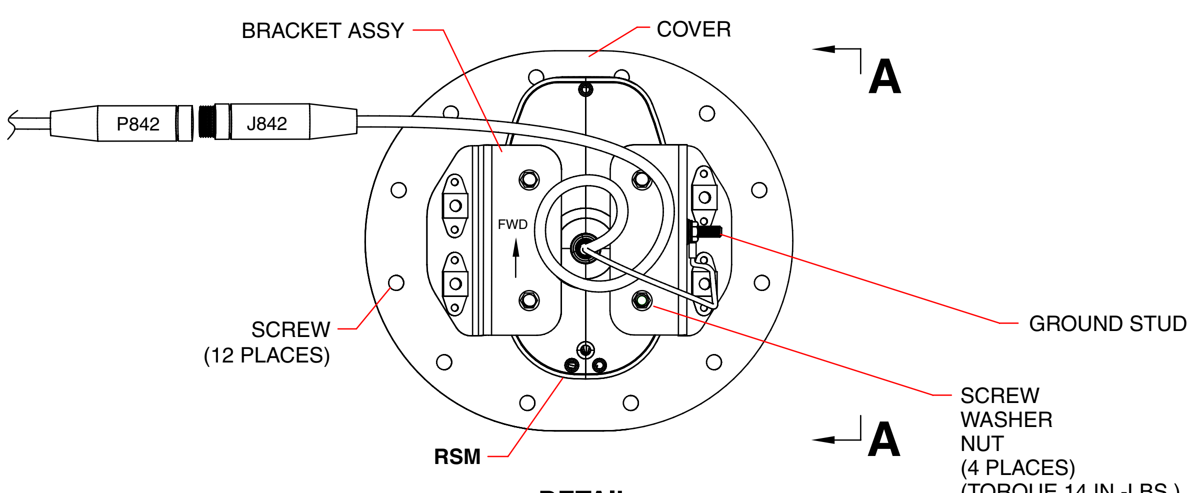


LEFT WING LOOKING UP



VIEW A - A
(LOOKING OUTBOARD)



DETAIL
(ROTATED 180°, LOOKING DOWN,
COVER REMOVED FROM AIRPLANE)

RSM Installation
Figure 2

(3) Configuration Module (CM)

The Configuration Module retains system configuration settings and calibration data. The Configuration Module connects to the EBD through a short fabricated harness and is fastened to the main wiring bundle of the display.

Each display has an associated Configuration Module that retains that display's aircraft specific configuration, calibration data and user settings. This permits the display hardware to be replaced without re-entering the installation settings or re-calibrating the EBD.

(a) Removal

Verify power is off. Cut the two (2) cable ties affixing the CM to the PFD wiring harness. Unplug the Molex connector by pressing down on the locking tab and gently pulling the connector from the module.

(b) Installation

Verify power is off. Plug the Molex connector into the module until it clicks. Cable tie the module to the PFD wiring harness.

(c) Return to Service

1) Configure unit per "Installation Menu – Unit Configuration" on page 342414.

2) Perform "RSM Calibration" on page 342424.

F. Post Installation Setup Procedure

(PIR-107920 New.)

(1) Configuration and Equipment Checkout

The following Test Equipment will be required to complete the remaining steps in the ground test procedure:

- Pitot Static Test Set
- Digital Multimeter
- Electrical Bonding Tester

(2) Bonding Check

(a) Verify braided bonding strap is installed between EBD ground stud and airframe ground at rear of instrument panel. Verfify EBD is bonded to airframe ground with no greater than 3 milliohms resistance.

(b) Verify EBD mounting bracket is bonded to instrument panel with no greater than 3 milliohms resistance.

(c) Verify RSM is bonded to RSM bracket assembly with no greater than 3 milliohms resistance.

(d) Verify RSM bracket assembly is bonded to access cover with no greater than 3 milliohms resistance.

(3) Initial System Turn On

(a) Apply ground power to airplane.

(b) Remove power by pulling applicable circuit breakers.

(c) Install the EBD, RSM, and Configuration Module, if required.

(d) Push in all applicable circuit breakers and apply power.

(e) Verify the EBD displays the initialization page after 5 seconds.

NOTE: AHRS Flags may take up to 3 minutes to clear. Airspeed and Altitude flags may take up to 20 minutes to clear at temperatures below -20°C.

(4) System Configuration

Configure the EBD system prior to running the ground test procedure. The configuration pages are accessed through the EBD display using the MENU button and the lower Right Control Knob labeled MODE/SYNC.

(a) Main Menu Access

The Main Menu operation is accessed by pushing the “MENU” button. See EBD Pilot Guide for Main Menu operation.

(b) Menu Navigation

When no fields are enabled for editing, rotating the right control knob clockwise advances to the next menu page and counterclockwise advances to a previous menu page.

Editable menu items are displayed in white text on a blue background, non-editable menus items are green text on a blue background while grey text on a blue background is disabled from editing.

(c) Edit Mode

Pushing the line select key adjacent to an editable field enables the associated field for editing. The field turns magenta when enabled and the right control knob reads “Edit Value”.

When the field is enabled for editing rotating the right control knob will adjust the value. Pushing the right control knob or the adjacent line select key will exit from the editable field.

(d) Main Menu Configuration

The Main Menu consists of pages that are pilot selectable. Refer to the EBD Pilot Guide for Main Menu configuration if necessary.

(e) Approved Software

Installation of approved software can be verified as follows:

1) Push the "MENU" key.
2) Rotate the right control knob clockwise until the system status page (see "Figure 3") appears.
3) Compare displayed software versions with "Chart 2" on page 342415.

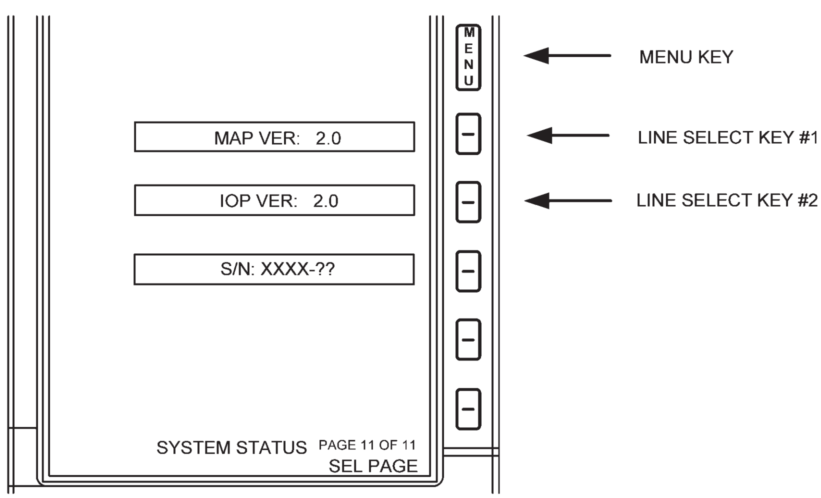
(f) Installation Menu – Unit Configuration

The Installation Menu is entered from the Main Menu's "SYSTEM STATUS PAGE" (last page). Simultaneously push and hold the MENU key, Line Select Key #1 and Line Select Key #2 for 3 seconds while the airspeed is below 30 units.

Whenever the warning message in "Figure 4" on page 342415 is displayed, pressing either control knob shall advance the Installation menu.

To exit the Installation Menu at any time press the MENU key. All data will be saved as displayed. The system will reboot and "INITIALIZING" will appear on the display for approximately 40 seconds.

Configure the EBD system per the following instructions.



Example Installation Menu Access
Figure 3

CHART 2
APPROVED SOFTWARE

Identifier	Version	Part Number
MAP	2.6.3	302-00007-015
IOP	2.0.5	302-00013-005

(PIR-107757 B.)

Installation Menu Warning
Figure 4

(g) EBD Installation Menu Configuration

Installation Date: ______________	
Aircraft Model: ________________	EFD1000 S/N: ______________
Aircraft Type: _________________	RSM S/N: _________________
Aircraft S/N: __________________	ACU S/N: _________________
	CM S/N: __________________

1) INSTALLATION MENU PAGE - IAS CONFIG A

Feature	Options	Actual Setting
MOL TYPE	1,2,3,4,5,6	1
MOL PARAM SEL	V_{ne}, V_{mo}, V_{no}, M_{mo}, MAX ALT	V_{ne}
MOL PARAM VALUE		198
MOL MARKER/BAND	BARBER POLE, REDLINE	REDLINE
MACH DSPL TYPE	ALTITUDE, SPEED	SPEED

2) INSTALLATION MENU PAGE – IAS CONFIG B - Set Speed Bands

Feature	Options	Actual Setting
OVERSPEED ALERT	DISABLE, ENABLE	DISABLE
V_{no}	0 to 450	168
V_{fe}	0 to 450	118
V_{s}	0 to 450	69
V_{so}	0 to 450	58

3) INSTALLATION MENU PAGE – IAS CONFIG C

Feature	Options	Actual Setting
V_{yse}	0 to 450	0
V_{mc}	0 to 450	0
Triangle	0 to 450	0
Not Used		
Not Used		

4) INSTALLATION MENU PAGE - IAS CONFIG D - Rotorcraft Only

5) INSTALLATION MENU PAGE - IAS CONFIG E - Color of Airspeed Tape Speed Bands

Feature	Options	Actual Setting
SPD Band 2	YELLOW, CLEAR	YELLOW
SPD Band 3	GREEN, WHITE, CLEAR	GREEN
SPD Band 4	WHITE, CLEAR	WHITE
Not Used		
Not Used		

6) INSTALLATION MENU PAGE – IAS CONFIG F

Feature	Options	Actual Setting
IAS UNITS	KTS, MPH	KTS
TAPES	UNLOCKED, LOCK OFF, LOCK ON	LOCK ON
VSPD EDIT	UNLOCKED, LOCKED	LOCKED
Not Used		
Not Used		

7) INSTALLATION MENU PAGE - IAS CONFIG G

Feature	Options	Actual Setting
IAS DISPLAY	DISABLE, ENABLE	DISABLE
ALT DISPLAY	DISABLE, ENABLE	ENABLE
BARO DISPLAY	DISABLE, ENABLE	ENABLE
VSI DISPLAY	ALWAYS ON, ALWAYS OFF, AUTO	ALWAYS ON
Not Used		

8) INSTALLATION MENU PAGE – NAV SETUP A

Feature	Options	Actual Setting
GPS / NAV #1	NONE,A,B,C,D,E,F,G,H,I,J,K,L,M,P,Q,R,S	NONE
GPS / NAV #2	NONE,A,B,C,D,E,F,G,H,I,J,K,L,M	NONE

9) INSTALLATION MENU PAGE – NAV SETUP B

Feature	Options	Actual Setting
429 IN PORT 1	NONE, GPS1, VLOC1, VLOC1+ACU, GPS1+ACU, GPS1+ACU+VLOC1, GPS2, VLOC2, VLOC2+ACU, GPS2+ACU, GPS2+ACU+VLOC2	NONE
429 IN PORT 2	Same as IN PORT 1 options	NONE
429 IN PORT 3	Same as IN PORT 1 options	NONE
429 IN PORT 4	Same as IN PORT 1 options	NONE
429 IN PORT 5	NONE, TRAFFIC	NONE

10) INSTALLATION MENU PAGE – NAV SETUP C

Feature	Options	Actual Setting
429 OUT PORT SPEED	HIGH, LOW	LOW
429 IN PORTS 1 & 2 SPEED	HIGH, LOW	LOW
429 IN PORTS 3 & 4 SPEED	HIGH, LOW	LOW
ADF CONFIG	NONE, 1, 2	NONE
NAV #2 POSITION SOURCE	GPS1, GPS2	GPS1

11) INSTALLATION MENU PAGE – RS232 PORT CONFIG A

Feature	Options	Actual Setting
232 IN PORT 0	NONE, RSM, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 1	NONE, GPS TYPE 1, GPS TYPE 2, GPS TYPE 3, GPS TYPE 4, GPS TYPE 5, XM, DFC A/P	NONE
232 IN PORT 2	NONE, GPS TYPE 1, GPS TYPE 2, GPS TYPE 3, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 3	NONE, WX500, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 4	NONE, EFD1000 MFD, EFD1000 PFD, EFD500 MFD	NONE

12) INSTALLATION MENU PAGE – RS232 PORT CONFIG B

Feature	Options	Actual Setting
232 IN PORT 5	NONE, EFD1000 MFD, EBD PFD, EFD500 MFD	NONE
232 OUT PORT 0	NONE, RSM, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 1	NONE, XM, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 2	NONE, EFD, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 3	NONE, WX500, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE

13) INSTALLATION MENU PAGE – RS232 CONFIG C
This page is grayed out and not configurable.

14) INSTALLATION MENU PAGE – ACU CONFIG A

Feature	Options	Actual Setting
ACU HSI Type	0, 1, 2, 3	0
ACU FD Type	0 thru 12	0
ACU Datum	NORMAL, REVERSED	NORMAL
FD Pitch Offset ADJ	-10.0 to +10.0 degrees	0.0
FD Roll Offset ADJ	-10.0 to +10.0 degrees	0.0

15) INSTALLATION MENU PAGE – ACU CONFIG B

Feature	Options	Actual Setting
Digital Nav Tx Speed	HIGH. LOW	LOW
Digital Nav Rx Speed	HIGH, LOW	LOW

16) INSTALLATION MENU PAGE – MISC CONFIG A

Feature	Options	Actual Setting
RSM Orientation	TOP, BOTTOM (Inverted orientation)	BOTTOM
RSM GPS Enable	DISABLE, ENABLE, INTERCOM	DISABLE
RSM GPS USAGE	EMER ONLY/ MODE 2	EMER ONLY
RAD ALT CONFIG	NONE, TYPE 1, TYPE 2, TYPE 3, TYPE 4, TYPE 5, TYPE 6, TYPE 7	NONE
WIND DISPLAY	DISABLE, ENABLE>=30KIAS, ENABLE>=40KIAS, ENABLE>=50KIAS, ENABLE>=60KIAS, ENABLE>=70KIAS, ENABLE>=80KIAS, ENABLE>=90KIAS	DISABLE

17) INSTALLATION MENU PAGE – MISC CONFIG B

Feature	Options	Actual Setting
ELEC SYSTEM	14 VOLT, 28 VOLT	14 VOLT*
EFD BATTERY CONFG:	INTERNAL, REMOTE	INTERNAL
ATTITUDE REF SYMBOL ADJ:	-5.0 to +5.0 degrees	0
PANEL TILT PITCH ADJ	-10.0 to +20.0 degrees	0
PANEL ROLL ADJ	-2.0 to +2.0 degrees	0

* Voltage at which the EFD1000 switches to internal battery.

18) INSTALLATION MENU PAGE – MISC CONFIG C

Feature	Options	Actual Setting
COMPOSITE PHASE (VOR1,2)	(0,0) (180,0) (0,180) (180,180)	0,0
GPSS GAIN:	0.5 to 2.0 (0.1 increments)	1.0
CRS SDI	COMMON, NAV 1/2	COMMON
OBS DISPLAY	DISABLE, ENABLE	DISABLE

19) INSTALLATION MENU PAGE – MISC CONFIG D

Feature	Options	Actual Setting
OAT SOURCE	NONE, RSM, PROBE, INTERCOM	NONE
OAT BIAS	0 to -8 degrees	0

20) INSTALLATION MENU PAGE – RSM CALIBRATION

The following menu will be used in the next section to calibrate and validate the magnetometer in the RSM. Heading errors of up to +/- 6.0° can be calibrated at 30° increments beginning with North. Follow "RSM Calibration" on page 342424.

Feature	Options
START CALIBRATION	Press to initiate
ACCEPT CALIBRATION?	Press to ACCEPT Cal
REJECT CALIBRATION?	Press to REJECT Cal
HDG SEL	030° to 360° (in 30° increments)
HDG ADJ	-6.0 to +6.0 (degrees)

Use the following table to record the HDG ADJ values used to bring the compass heading values into specification:

Rsm Calibration	Page 12	Options	Actual Setting
HDG SEL: 030	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 060	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 090	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 120	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 150	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 180	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 210	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 240	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 270	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 300	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 330	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________
HDG SEL: 360	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	____________

21) INSTALLATION MENU PAGE – ACCEL BIAS CAL
This page is for Factory Calibration only.

22) INSTALLATION MENU PAGE – WX-500

Feature	Options	Actual Setting
HEAD TYPE	NONE, DISPLAY, CONTROL	NONE
ANTENNA MOUNT	TOP, BOTTOM	NONE

23) INSTALLATION MENU PAGE – NETWORK PAGE
This menu is for diagnostic purposes only.

Feature	Options
IP ADDR	0-255
SUBNET MASK	0-255
GATEWAY	0-255
PORT	0-9999
Not Used	

Notes: Repeated presses of top 3 line select keys will select one of 4 selectable fields

24) INSTALLATION MENU PAGE - DIAGNOSTICS

This menu is for diagnostic purposes only. See "Figure 5".

NOTE: Operational Time and Flight Time are stored in the configuration module of the aircraft and these times are cumulative for the installation not the EBD unit itself.

25) EXITING / SAVING DATA

To exit the Installation Menu press the MENU button. All data will be saved as it was displayed on each page. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds. Normal operation continues.

Diagnostics
Figure 5

(h) IAS Configuration

1) Definitions

V_{ne}: Never Exceed Speed (redline)

V_{no}: Maximum Structural Cruise Speed (bottom of yellow band)

V_s: No Flap Stall speed (bottom of green band)

V_{fe}: Maximum Flap Extend speed (top of white band)

V_{so}: Full Flap Stall speed (bottom of white band)

V_{yse}: Single Engine best rate of climb (blue marker)

V_{mc}: Single Engine minimum control speed (red marker)

◁ Initial Flap Extension Speed

2) Use the following table to fill in Configuration Menu Page 1:

INDICATED AIRSPEEDS (IAS) CONFIGURATION

IAS Setting (KIAS)	Band Color	Band Range	Description	Pass
V_{ne} = 198	Red	= V_{ne}	Beginning of Red Band	_____
V_{no} = 168	Yellow	= V_{no}	Bottom of Yellow Band	_____
V_s = 69	Green	V_s - V_{no}	Green arc extending from V_s to V_{no}	_____
V_{fe} = 116	White	V_{so} - V_{fe}	Top of White arc	_____
V_{so} = 58	White		Bottom of White Arc	_____
V_{yse} = 0	N/A		Set to zero for single engine airplanes	_____
V_{mc} = 0	N/A		Set to zero for single engine airplanes	_____
◁ = 165	Triangle (White)	= ◁	White triangle at initial flap extension airspeed	_____

3) ATTITUDE REF SYMBOL ADJ

This is used to make small changes to the pitch attitude reference mark on the EBD should the aircraft fly consistently slightly nose high or nose low in cruise trim. It is not necessary to perform the RSM Calibration after making changes to this setting.

4) EXITING / SAVING DATA

To exit the Installation Menu press the MENU button. All data will be saved as it was displayed on each page. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds. Normal operation continues.

(5) RSM Calibration

(a) Calibration Overview

The Remote Sensor Module must be calibrated by performing a compass swing in the aircraft for any maintenance activities that could affect RSM accuracy.

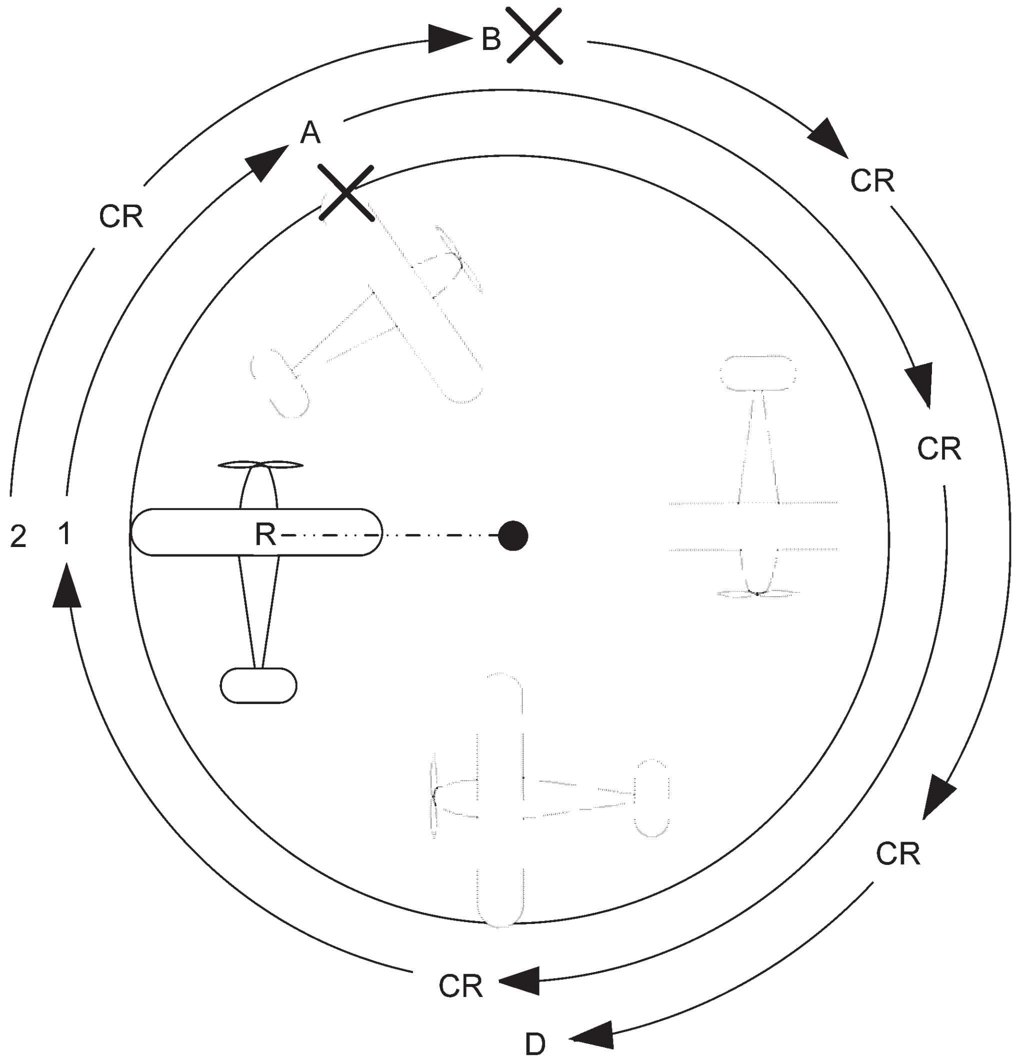
Such activities might include but are not limited to the replacement of the RSM, replacement of the Configuration Module, installation of a mechanical or electrical device in the vicinity to the RSM, installation of an appliance that might generate a magnetic interference.

NOTE: Before replacing the RSM and/or the Configuration Module, determine if the current installation has had Aspen SB2009-02 applied. Refer to the EBD Configuration Chart (if available) or the airplane logbooks. If SB2009-02 has been applied, do not replace either the RSM or the Configuration Module without completing the calibrations required by SB2009-02.

An overview of the RSM Calibration procedure is as follows (see "Figure 6" on page 342425):

CAUTION: WHEN A CALIBRATION IS INITIATED, THE AIRCRAFT MUST BE TURNED AS DESCRIBED BELOW. IF THE CALIBRATION IS INITIATED AND THEN ACCEPTED WITHOUT MOVING THE AIRCRAFT, AN ERRONEOUS CALIBRATION WILL BE WRITTEN TO THE CONFIGURATION MODULE, POTENTIALLY RESULTING IN FAILURE TO INITIALIZE.

- A heading will be checked to verify the reasonableness of EBD heading prior to calibration.
- The aircraft will be taxied to a magnetically quiet and level area at least 200ft from metal buildings and clear of metal grates, manhole covers and rebar within the concrete. A Compass Rose is ideal for this procedure.
- The aircraft can start from any heading.
- With engines running, all electrical equipment operating, and the aircraft stationary the RSM CAL page will be entered and Start Calibration will be initiated (see "Figure 7" on page 342426).
- After a 15 second countdown timer the pilot/operator will begin to taxi the aircraft in a circle (cw or ccw) with the radius of approximately twice the length of the aircraft's wing as viewed from the cockpit (≈ 30ft).
- The aircraft will be taxied under its own power at a constant rate around a circle until a 60 second timer elapses. The aircraft must not stop until the timer has exhausted.
- At the completion of the 60 seconds the aircraft will have made at least a 450° circle (360° + 90°) to approximately two complete circles (720°).
- At the end of the 60 second timer four headings about 90° apart will be checked against a calibrated heading source (i.e., site compass, compass rose).
- If EBD heading is acceptable then the calibration is Accepted.
- If the EBD heading is not within tolerance then it is Rejected and the calibration procedure is re-run.
- After the calibration is accepted headings are checked using a calibrated reference (i.e., a sight compass) every 30° (starting from North) to verify that the heading accuracy is within ±4°.
- Heading can be calibrated every 30° if necessary.



●	=	Imaginary pivot point
R	=	Radius to pivot point – approximately equal to wingspan (≈30ft)
1	=	Starting Point
A	=	Established Constant Rate turn ≈ 10 seconds after 1
CR	=	Constant Rate Turn must continue until end of 60sec timer
2	=	360° turn complete – continue at Constant Rate – Do not stop
B	=	60sec timer must not reach Zero before this point
D	=	Accept/Reject CAL message occurs between B & D

RSM Calibration Graphic
Figure 6

(b) Calibration Procedure

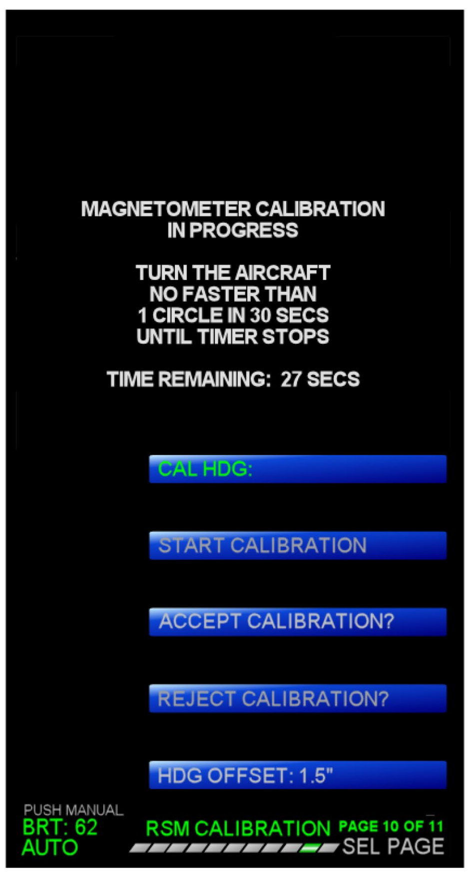
The RSM calibration routine is accomplished using the Installation Menu “RSM CALIBRATION” menu page (see "Figure 7"). See "Installation Menu – Unit Configuration" on page 342414 for instructions on entering the INSTALLATION MENU.

On the “RSM CALIBRATION” menu page the current calibrated heading (to the nearest 0.1 degrees) will be continuously displayed adjacent to the “CAL HDG:” menu field.

NOTE: Verify CAL HDG is within 25 degrees of a calibrated compass source prior to starting the RSM Calibration. This is a rough check of the hard and soft iron effects (magnetic interference) in the vicinity of the RSM. Should the CAL HDG be greater than 25 degrees then the RSM location should be re-surveyed.

With aircraft stationary at (POSITION 1) of "Figure 6" on page 342425 – RSM Calibration Graphic press the “START CALIBRATION” line select key. The annunciation shown in "Figure 8" on page 342427 will be displayed with a countdown timer that begins with 15 secs and counts down to 0 secs.

CAUTION: DO NOT PRESS “ACCEPT CALIBRATION” WITHOUT MOVING THE AIRCRAFT IN THE PROCEDURE BELOW AS CORRUPTION TO THE CONFIGURATION MODULE COULD OCCUR.



RSM Calibration Page
Figure 7

MAGNETOMETER CALIBRATION
IN PROGRESS

DO NOT MOVE THE AIRCRAFT
FOR THE NEXT 7 SECS

Calibration in Process
Figure 8

MAGNETOMETER CALIBRATION
IN PROGRESS

TURN THE AIRCRAFT
NO FASTER THAN
1 CIRCLE IN 30 SECS
UNTIL TIMER STOPS

TIME REMAINING: ## SECS

Aircraft Turning
Figure 9

When the menu of "Figure 9" is displayed immediately begin taxiing the aircraft clockwise or counter-clockwise at a constant rate of no faster than 1 turn every 30 seconds. About half normal taxi speed or a brisk walking speed is about right. Approximately 10 seconds after initial movement (see POSITION A) the aircraft should be taxiing at a constant rate (CR) throughout the rest of the procedure. When the countdown timer is reached between one and a quarter turns (450°) (POSITION B) and two turns (720°) (POSITION D) should have been completed. While turning the aircraft do not stop the aircraft until the end of the 60 second timer and "Figure 10" appears.

MAGNETOMETER CALIBRATION
COMPLETE

PLEASE ACCEPT OR REJECT RESULTS

Accept/Reject Results
Figure 10

If you find that the timing of the turns was not right such that "Magnetometer Calibration Complete" message occurs between B & D of "Figure 6" on page 342425, then REJECT the results and re-run the procedure.

At the end of the calibration routine the "ACCEPT CALIBRATION?" and "REJECT CALIBRATION?" menu options will be enabled.

NOTE: If the message "Erroneous Calibration Values" is displayed then magnetic interference exists in the vicinity of the calibration area or the RSM is mounted in a magnetically noisy area. Try the calibration process again in a flat magnetically quiet area. If the message is displayed again the RSM location must be surveyed for magnetic interference.

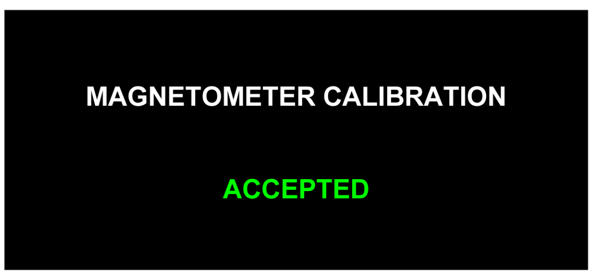
To determine whether to ACCEPT or REJECT the results check four headings approximately 90° apart against a known good heading source (i.e., aircraft compass, sight compass, compass rose).

If the headings are within ±10° then press ACCEPT and use the Heading Offset Adjustment, below, to align each heading value to actual.

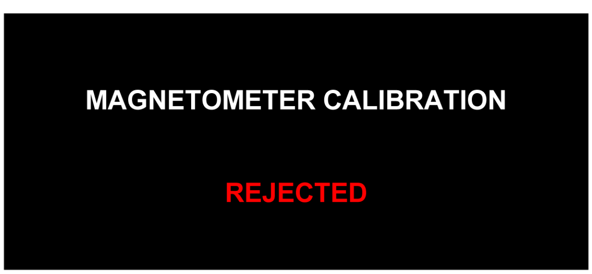
NOTE: Typically with a clean RSM location the heading values will be within 3 degrees of actual heading. If they are outside this value the RSM mounting area should be rechecked for magnetic contamination (internal interference) or the calibration can be rerun in a new location (external interference) to see if the values are improved. RSMs tilted to the upper end of the limit may not be able to achieve the 3 degree or less of error without using the Heading Offset Adjustment).

Pressing the "ACCEPT CALIBRATION" selection shall accept the calibration results, display the annunciation shown in "Figure 11" for 5 seconds, and return the "RSM CALIBRATION" menu page to its initial state.

Pressing the "REJECT CALIBRATION" selection shall reject the calibration results. Reject the results if the calibration was poor or a previously stored calibration has better heading accuracies.



Results Accepted
Figure 11

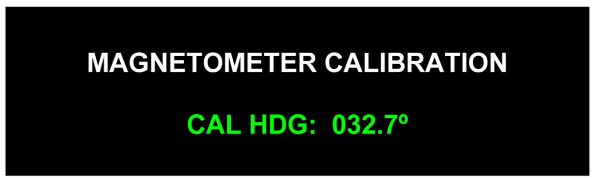


Results Rejected
Figure 12

(c) Heading Offset Adjustment

When the calibration routine is complete and the results have been accepted the CAL HDG value will be displayed.

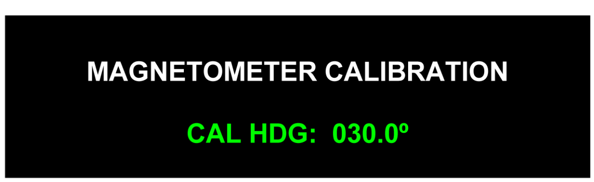
Position the aircraft so that it is at a heading of 30° as verified by a sight compass or other calibrated means. Press the line select key next to “HDG SEL” and then turn the right knob until HDG SEL: 030° is displayed. Now press the line select key next to “HDG ADJ” and turn the right knob until the CAL HDG of "Figure 13" displays exactly 030.0° as shown in "Figure 14" or as close to 030° as possible within a ±4° tolerance. The HDG ADJ field will display the amount of correction (+/- 6.0 degrees) that was required. Press the HDG ADJ line select key to accept the setting.



Calibration Heading before adjustment
Figure 13

Now repeat the process above for all other headings in 30° increments from 60° to 360°. Press the MENU key to exit the Installation Menu. Continue with heading accuracy tests below.

NOTE: In some aircraft, prop wash and wind during ground operations can create inconsistent pressures in the pitot-static system. The pressures can affect the ADAHRS, resulting in small pitch and heading perturbations. Before reading the aircraft headings for the purposes of calibration, ensure the attitude solution has stabilized and is not influenced by external winds and pressures. The disturbances normally settle out over a period of 15 to 60 seconds. Idle power or temporarily selecting the alternate static source can sometimes eliminate the effect.



Calibration Heading after adjustment
Figure 14

(d) Heading Accuracy Test

As a final check, position the aircraft on the headings shown in the **HEADING ACCURACY TEST** table below and verify heading (viewed on HSI display) is within +/- 4 degrees using a calibrated heading source (i.e., sight compass, compass rose).

If any heading is outside ±4° then rerun the Calibration Procedure and or Heading Offset adjustment.

HEADING ACCURACY TEST

Calibrated Heading Source	Tolerance	Actual EBD Heading	Calibrated Heading Source	Tolerance	Actual EBD Heading
30	+/- 4	___________	210	+/- 4	___________
60	+/- 4	___________	240	+/- 4	___________
90	+/- 4	___________	270	+/- 4	___________
120	+/- 4	___________	300	+/- 4	___________
150	+/- 4	___________	330	+/- 4	___________
180	+/- 4	___________	360	+/- 4	___________

(e) Heading Interference Test

With aircraft engine(s) running monitor current aircraft heading on EBD and exercise flight controls stop to stop including flaps and any electric trim tabs. Verify the heading (viewed on HSI display) does not change on the EBD by more than 2°. If movement of flight controls causes more than a 2° heading change then it may be necessary to degauss the flight controls including the cables. A handheld degausser can be found at most audio and video stores.

Operate all electrical and environmental equipment including:

- Blowers, fans, heaters, air conditioner
- Deice boots, fuel pump(s), backup vacuum pumps
- Landing, recognition, strobe, NAV lighting
- Operate pulse equipment – transponder, WX radar, DME
- Key all VHF communication radios.
- Operate autopilot so that all servos run (roll, pitch, yaw, trim)

If the operation of any electrical system causes the heading to change by more than 2° the RSM wiring may need to be relocated away from the offending system. The offending system may also have a bonding issue to the airframe that needs to be corrected.

Run engine(s) from idle to take off power and verify that the heading does not change by more than 2°. Prop wash at higher engine RPMs may cause a heading shift, try an alternate static source if this is a issue.

This completes all RSM calibration and heading tests.

(6) Ground Test Procedure

CAUTION: DO NOT EXCEED THE AIRCRAFT'S MAXIMUM AIRSPEED, ALTITUDE, OR VERTICAL SPEED AT ANYTIME DURING THE TESTING. DAMAGE COULD RESULT TO THE PRE-EXISTING AIRCRAFT INSTRUMENTS.

NOTE: When changing indicated airspeed or altitude on the ground using pitot/static test equipment, changes in the AHRS display of pitch, roll and heading will result, possibly accompanied by a CROSS CHECK ATTITUDE annunciation. This behavior is the result of the Kalman Filter algorithms employed in the EBD attitude solution. These changes in pitch, roll or heading are normal and do NOT indicate a system failure. The integrated nature of the EBD AHRS algorithms is such that AHRS performance can only be properly evaluated during flight or ground maneuvers.

(a) Check the following items for proper operation:

1) Airspeed Tape, Altitude Tape
2) AHRS Sensor
3) EMI Test
4) Flight Control Interference Check

(b) Indicated Airspeed Display

WARNING: THIS TEST MUST BE ACCOMPLISHED BY AN APPROPRIATELY RATED MECHANIC OR REPAIRMAN.

Use the Indicated Airspeeds (IAS) Configuration table under IAS Configuration. Set the Pitot/Static test set for 5000 ft above field elevation. Increase airspeed to V_{ne} and check all applicable Speed Bands and Speed Markers listed in table.

(c) Altitude Display

With the Pitot/Static tester still set for 5000 ft above field elevation and with BARO Set to 29.92 in. Hg. on the EBD, verify altitude tape displays altitude within ±40ft of the calibrated test set altitude.

(d) System Leak Test

Perform the pitot-static system leak "Test" on page 341014 or set the Pitot Static Test Set to 1000 ft above field elevation and without additional pumping for a period of 1 minute the aircraft static system should not lose more than 100 ft of altitude in a non-pressurized airplane.

(e) AHRS Sensor Test

Verify that correct aircraft attitude information is presented on the Attitude Indicator portion of the EBD. The Flags may take up to 5 minutes to clear when the ambient temperature is below -20° C. Typically the attitude solution will be available in less than 3 minutes.

(f) EBD Internal Battery Test

(PIR-900-00012-001, Rev. T.)

Test the internal back-up battery in the EBD every 12 months. Replace the battery if it fails the following operational test.

1) Setup

This test must be run at room temperature approximately 25° C.

2) Procedure

a) Apply external (ground) power.

b) Turn on the EBD

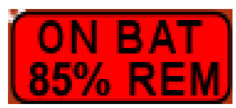
c) Press MENU Key

d) Select POWER SETTINGS page from the Main Menu

e) Press the BATTERY line select key

f) BAT LEVEL IN --.-- will be displayed for a short period of time as battery capacity is being measured. This could take up to 10 minutes if the ambient temperature is below 0° C.

g) Once the capacity is measured ON BAT XX% REM will be displayed.



h) The “ON BAT” indication must read a minimum of 80% to continue. If the battery capacity is below 80% then the battery should be charged by returning the EBD to external power. The battery will charge as long as the EBD is turned on and external power is supplied.

i) With the battery displaying greater than 80% charge set a timer for 30 minutes. After the 30 minute time has elapsed the EBD must still be operating on battery. If the internal battery will not supply the minimum 30 minutes operating time or fails to charge above 80%, replace the battery and return the failed battery to Aspen Avionics.

j) Instructions for battery replacement are under Components, EBD Display, above.

k) Switch the EBD back to external power and recharge the internal battery to 80% or greater prior to release to service.

(g) EMI Test

Monitor the EBD for Flags, Red-X's, Red Slashes, heading changes, altitude changes, airspeed changes, attitude changes or any error messages while performing the following Test:

1) Transmit on all Comm radios for 20 seconds each at 118.000MHz, 126.900MHz, and 136.950MHz.
2) Turn on all transponders, DMEs, Wx Radar, and all other pulse type equipment for 20 seconds each.
3) Operate all aircraft lighting including position lights, strobe lights, navigation lights, and all other forms of lighting for 20 seconds each.
4) Operate all environmental equipment including fans, air conditioning, heaters, and all other forms of environmental control equipment for 20 seconds each.
5) Operate fuel pump(s), deice boots, windshield heat, prop heat, flaps.
6) Operate autopilot, servos, yaw damper, and all electric trim servos.
7) Operate engine and verify no interference.

(h) Flight Control Interference Check

CAUTION: VERIFY CONTROL SURFACES ARE FREE AND CLEAR.

With the EBD installed in the instrument panel:

1) Push the control wheel all the way forward (nose down) and verify there is sufficient clearance between the EBD and its knobs, and the control wheel. With the control wheel fully forward move it from lock to lock (full right to full left) and verify there continues to be sufficient clearance between the EBD and its knobs, and the control wheel.
2) Pull the control wheel all the way aft (nose up) and verify there is no interference caused by the control column, rollers, arms, elevator torque tube, etc. at the rear of the EBD. With the control wheel fully aft move it from lock to lock (full right to full left) and verify there continues to be sufficient clearance at the rear of the EBD between it, its hoses and connnections, and any part of the control column and its associated parts and linkages.

(7) Post Installation Flight Check

CAUTION: ONLY PERFORM FLIGHT TEST IN DAY VFR CONDITIONS WITH AN APPROPRIATELY RATED PILOT FOR THE AIRCRAFT TYPE TO BE FLOWN.

(a) Basic ADI Flight Checks

Fly the aircraft in straight and level flight and verify that the ADI roll indication is level with reference to the horizon. Observe the Slip Indicator is centered under the Roll pointer and adjust rudder trim if available to center.

1) Make a coordinated 30 degree banked turn to the right and verify that the ADI roll indication is correct with reference to the horizon.

2) Make a coordinated 30 degree banked turn to the left and verify the ADI roll indication is correct with reference to the horizon.

3) Pitch the aircraft up 10 degrees and verify the ADI pitch indication is correct with reference to the horizon.

4) Pitch the aircraft down 10 degrees and verify the ADI pitch indication is correct with reference to the horizon.

(b) Basic HSI/DG Flight Checks

1) Make a 180 degree coordinated turn to the right and verify that the compass scale and numerical heading indication correctly track the aircraft heading during the turn.

2) Make a 180 degree coordinated turn to the left and verify that the compass scale and numerical heading indication correctly track the aircraft heading during the turn.

3) Then turn from West to North (30° Angle of Bank) and, using an outside reference, roll out to a northerly heading. (In the Southern Hemisphere, also turn West to South)

4) Immediately after the aircraft rolls out, record the heading indication.

5) Maintain the same heading by outside visual reference. There may be some movement of the heading indicator as the heading system stabilizes.

6) When the heading stabilizes, record the heading again. Then perform the test from East to North (In the southern Hemisphere, also turn east to South).

7) If the difference between the heading on rollout and the heading after stabilizing is more than 7°, refer to the Troubleshooting Chart above.

(c) Document Test Flight

This completes the flight test. If everything was satisfactory then document the completion of the Test Flight in aircraft log book in accordance with FAR 91.407(b).

GARMIN GI 275 STANDBY INSTRUMENT

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Garmin GI 275 Standby Instrument

Installed as standard equipment in S/N's 4636776 and up.

A. Description

The Garmin GI 275 is a multi-functional touchscreen instrument that is installed as a standby instrument displaying Attitude, Altitude, Airspeed, OAT, Heading, and fits in a standard 3-1/8-inch instrument cutout. The semi-round display accepts user input through the use of the capacitive touchscreen display and dual inner and out rotatory knobs on the bottom left of the instrument. The GI 275 is powered by the 7.5 amp standby instrument circuit breaker and has an internal lithium-Ion battery used to supply emergency power to the instrument in the event of an electrical failure for up to 60 minutes.

The GI 275 contains integrated air data, altitude and heading sensors to provide a secondary display of information. Additional equipment is installed to support the display including an external gps antenna and external magnetometer.

The GI 275 receives a GPS input for attitude aiding via the glareshield mounted gps antenna and provides ground track and ground speed (GS) functionality. Heading data for the GI 275 is provided from the remotely mounted GMU 11 Magnetometer. The pitot / static system is connected to the GI 275 and provides the following functions: Secondary Airspeed Indicator, Secondary Barometric Altimeter, Secondary Vertical Speed, Secondary Altimeter Barometric Setting, Secondary Altimeter, Secondary Altitude Display, Selected Altitude setting, bug, and altitude alerting, Secondary Vertical Speed Indicator, and Vspeed references.

Garmin GI 275 Standby Instrument
Figure 1

B. System Files and Logs

(1) Configuration File and Log

Prior to any maintenance being performed on the GI 275, the system configuration file must be saved to a USB drive and the system configuration log must be printed out. Both must be included with the permanent aircraft maintenance records.

To download and print the system configuration log, perform the following procedure:

(a) Power on the GI 275 in Configuration mode (see "Configuration Mode" on page 34240122).

(b) Insert a USB drive into the USB dongle. A USB icon should appear on the left of the display once the GI 275 has recognized the device. If the icon doesn't appear after 1 minute, remove the drive and re-insert it.

(c) Navigate to the Config Options page (SW/Config › Config Options).

(d) Touch the Export Config button.

(e) Touch the Select Name field and enter a name for the saved file.

(f) Touch the Export Config button.

(g) Once the save process is completed, disconnect the USB drive and insert it into a computer.

(h) On the computer, navigate to the USB drive and open the “summary” directory.

(i) Open the file with the name you entered to save your aircraft configuration.

(j) Print the configuration log.

(2) Maintenance Log

The GI 275 has a maintenance and error log that can be accessed or printed to assist with system maintenance and troubleshooting. The maintenance log will display reported system faults. If any faults are reported in the maintenance log, refer to "Troubleshooting" on page 3424013.

To download and print the system configuration log, perform the following procedure:

(a) Power on the GI 275 in Configuration mode (see "Configuration Mode" on page 34240122).

(b) Insert a USB drive into the USB dongle. A USB icon should appear on the left of the display once the GI 275 has recognized the device. If the icon doesn't appear after 1 minute, remove the drive and re-insert it.

(c) Navigate to the Maintenance page.

(d) Press the Export Logs button.

(e) Select Fault Log.

(f) Press the Start Download button.

(g) Once the save process is completed, disconnect the USB drive and insert it into a computer.

(h) On the computer, navigate to the USB drive and open the “maintenance_logs” directory.

(i) Open the .htm file.

(j) The file should open in your computer's default Internet web browser and can be printed using your browser's print function (in most cases, pressing Ctrl + P buttons simultaneously will access this function).

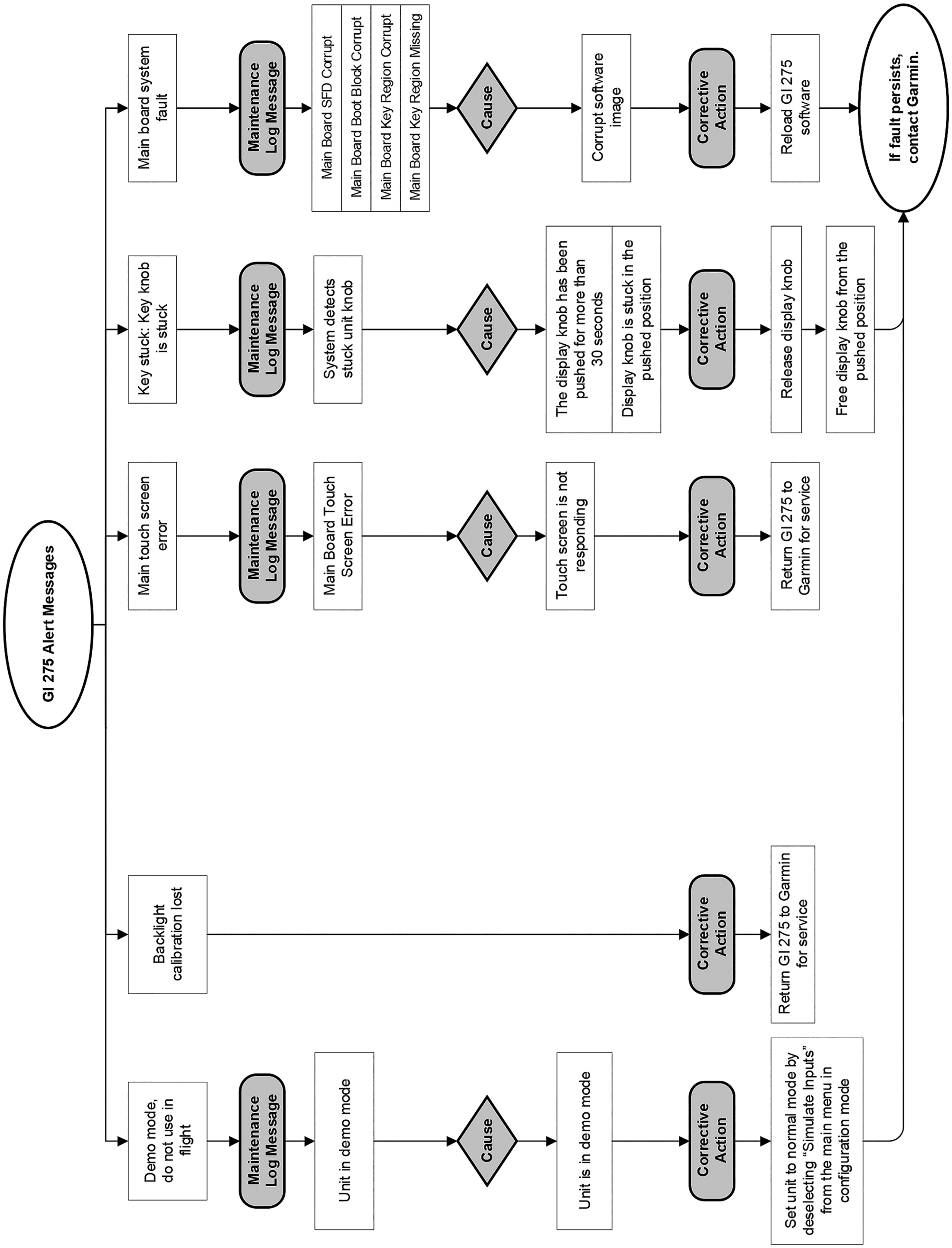
2. Troubleshooting

NOTE: Before troubleshooting, print the current configuration log using the procedure in "Configuration File and Log" on page 3424012 to ensure that system configuration settings match those recorded in the aircraft's permanent records as applicable to the aircraft's current configuration. If the GI 275 configuration does not match the configuration log retained with the aircraft permanent records, load the saved configuration from the USB drive retained with the aircraft records. If this cannot be accomplished, or does not correct the configuration, the issue must be corrected by a Garmin dealer.

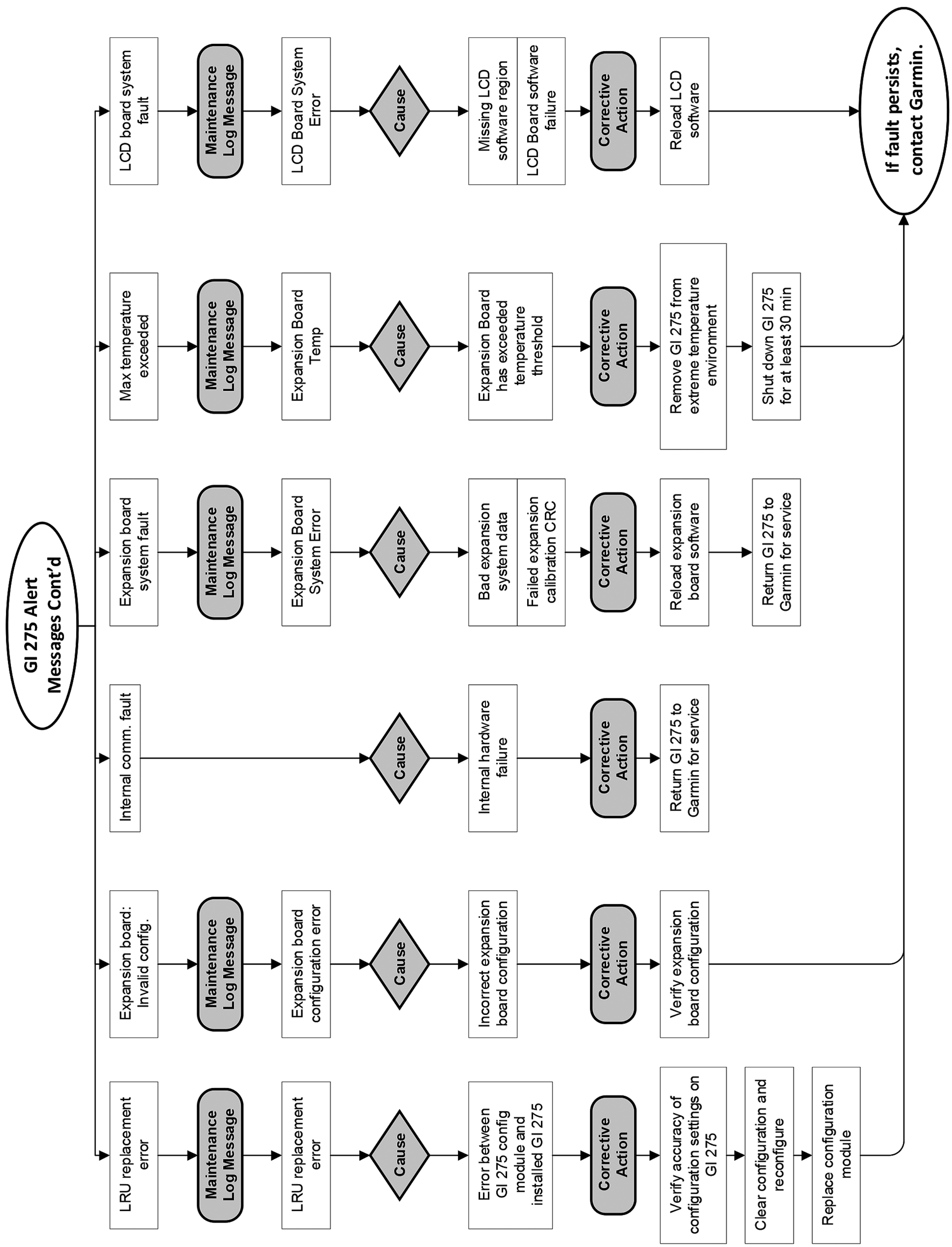
The Garmin GI 275 and system LRU's are designed to detect internal failures. A thorough self-test is executed automatically upon applying power to the units, and built-in tests are continuously executed while the LRU's are operating. Detected errors are indicated on the GI 275 display via failure annunciations, system messages, or a combination of the two.

A list of reported errors can be printed from the System Maintenance Log to assist in troubleshooting. (See "Maintenance Log" on page 3424012). Troubleshooting of the GI 275 can be accomplished using the chart below:

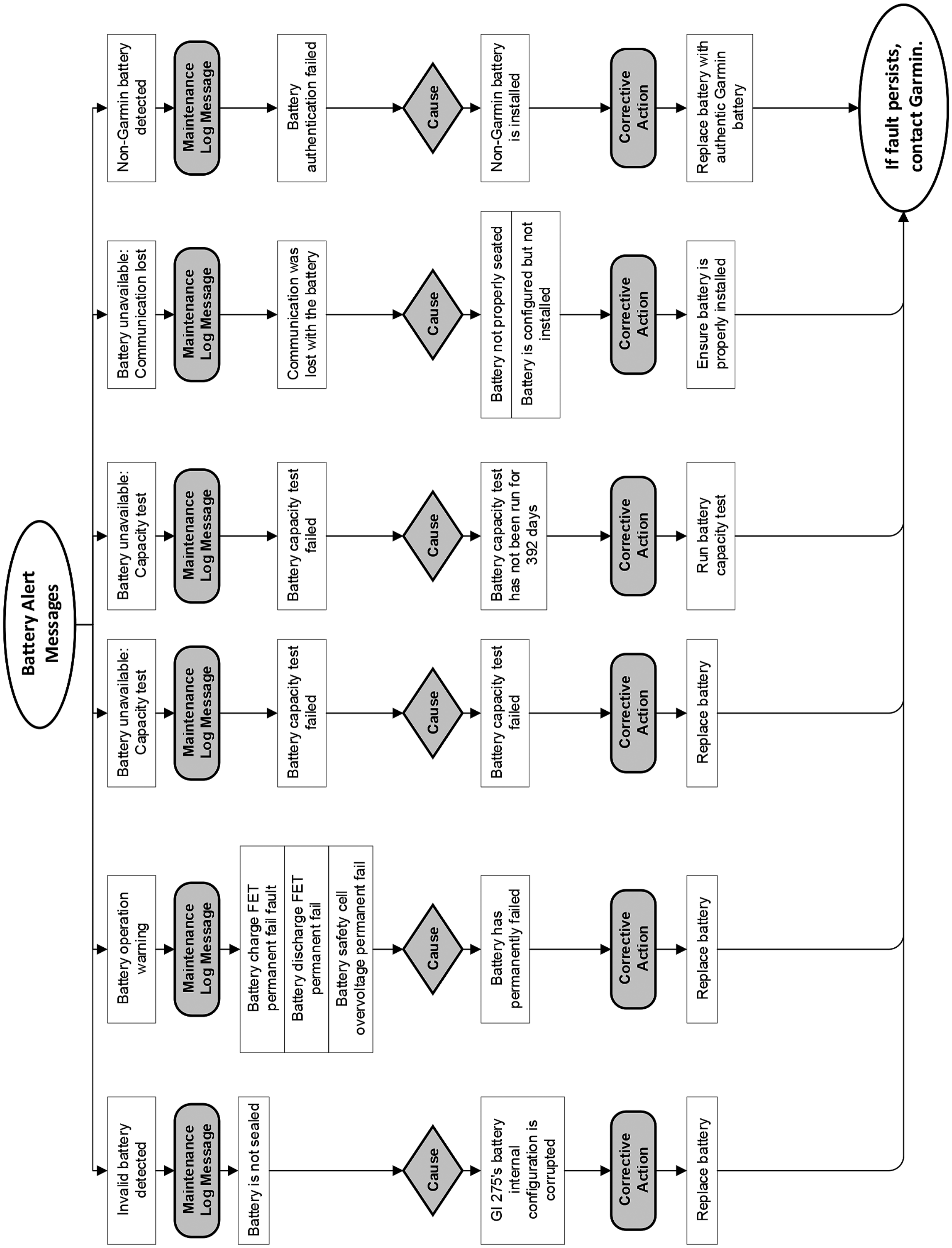
Symptoms	Recommended Action
The GI 275 screen is blank.	Check power/ground wiring for GI 275.
An alert message is displayed on the GI 275 or present in the maintenance log.	Troubleshoot the alert message using the flowcharts in "Figure 2" on page 3424014.



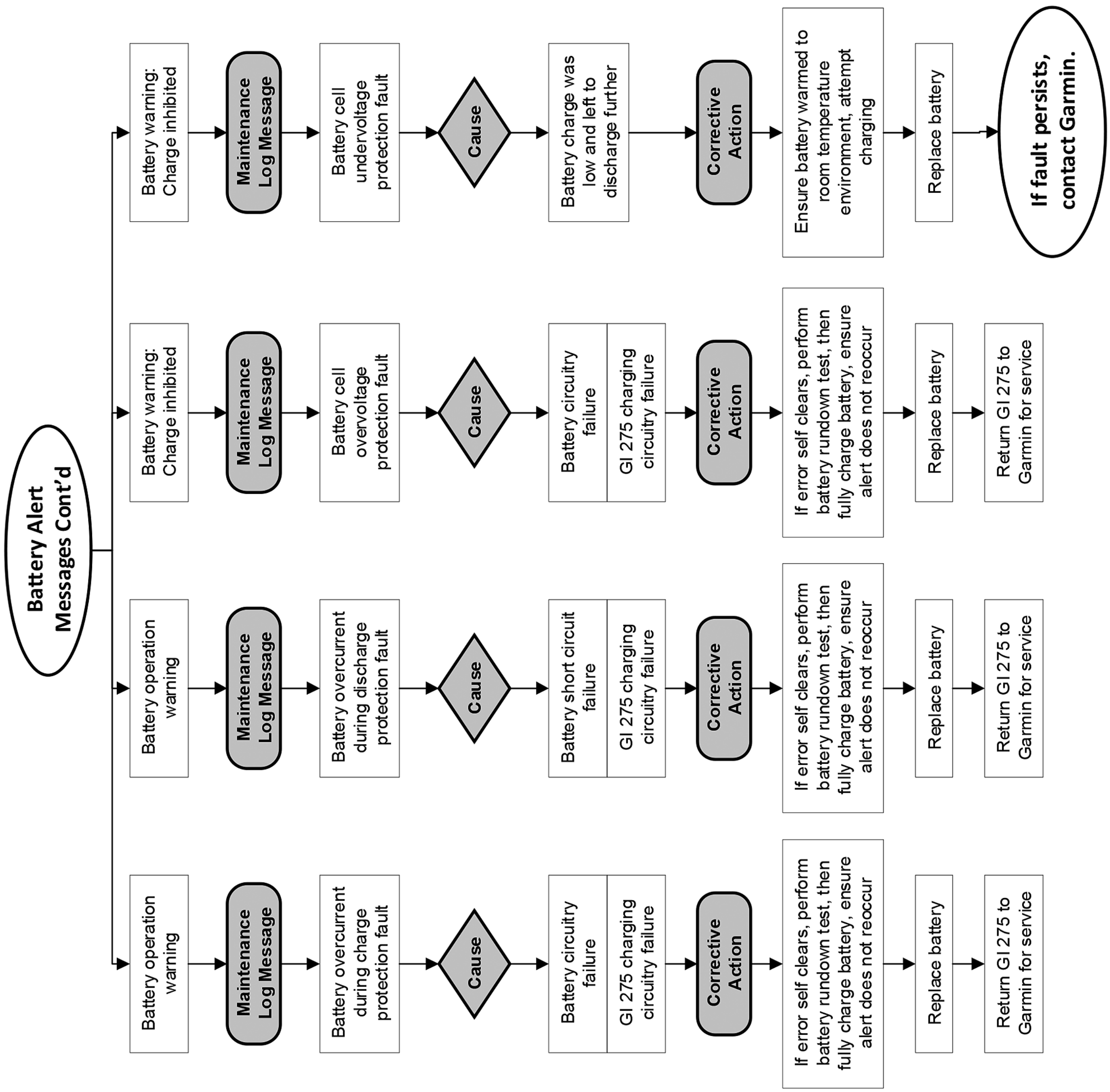
GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 1 of 9)



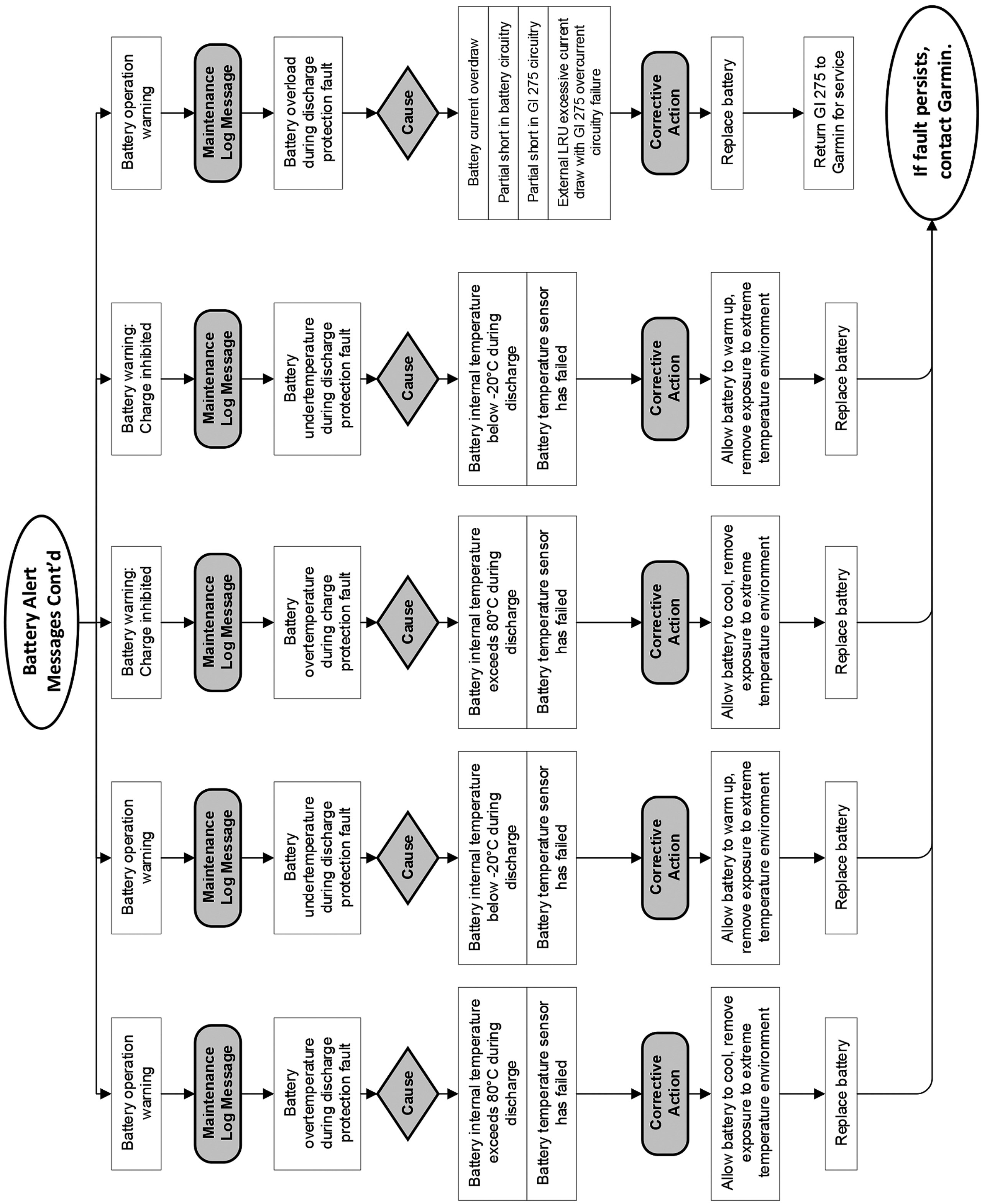
GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 2 of 9)



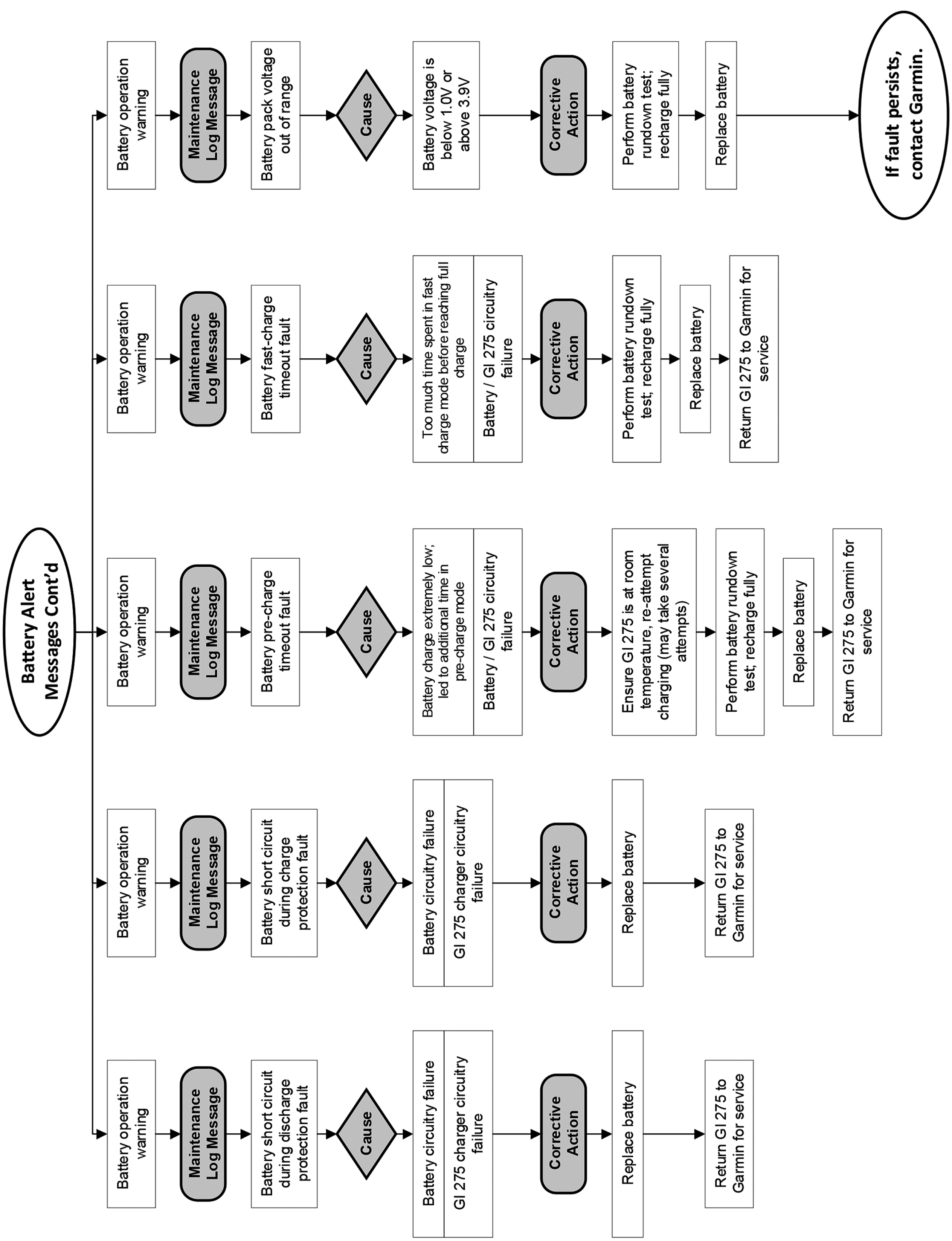
GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 3 of 9)



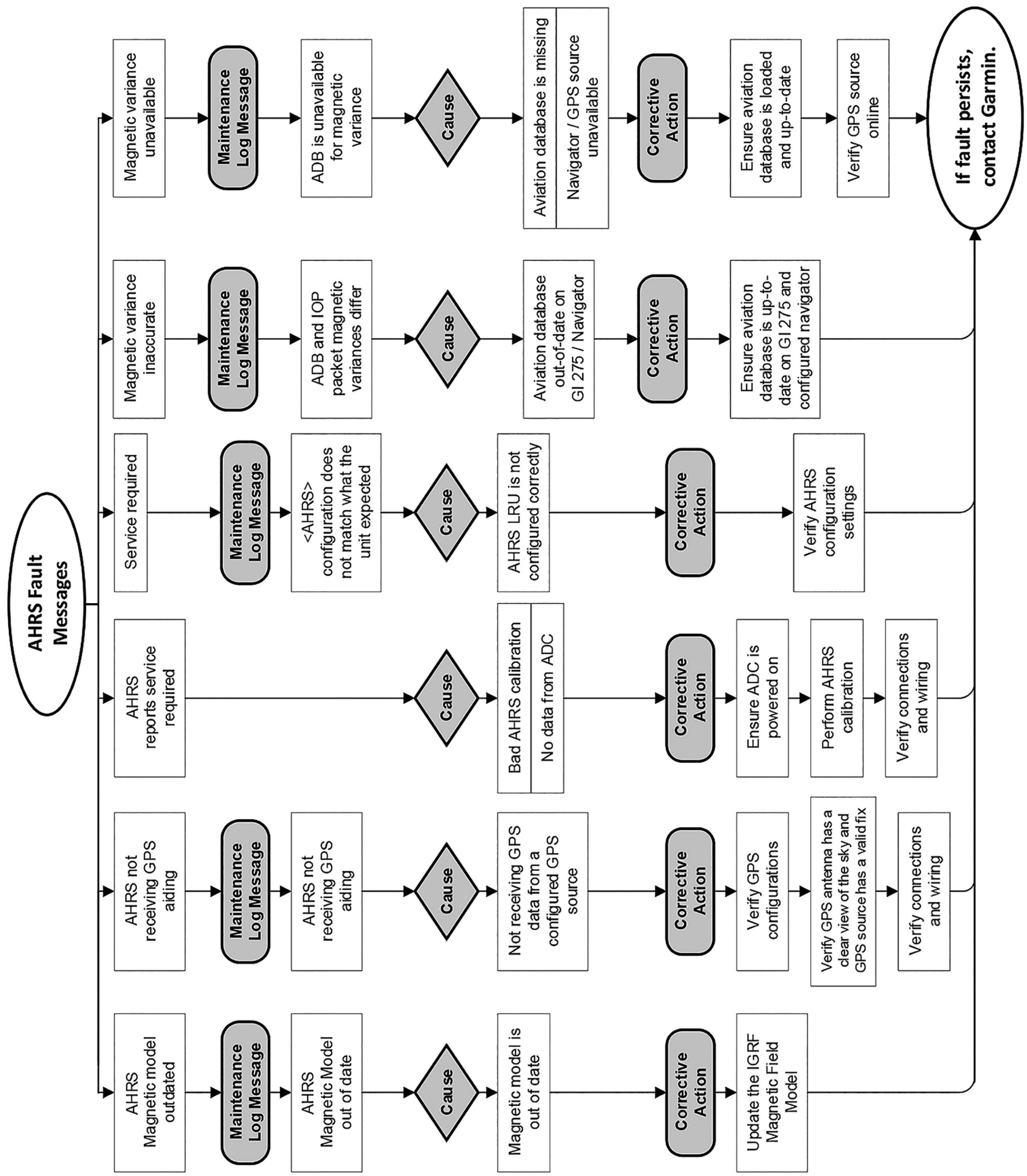
GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 4 of 9)



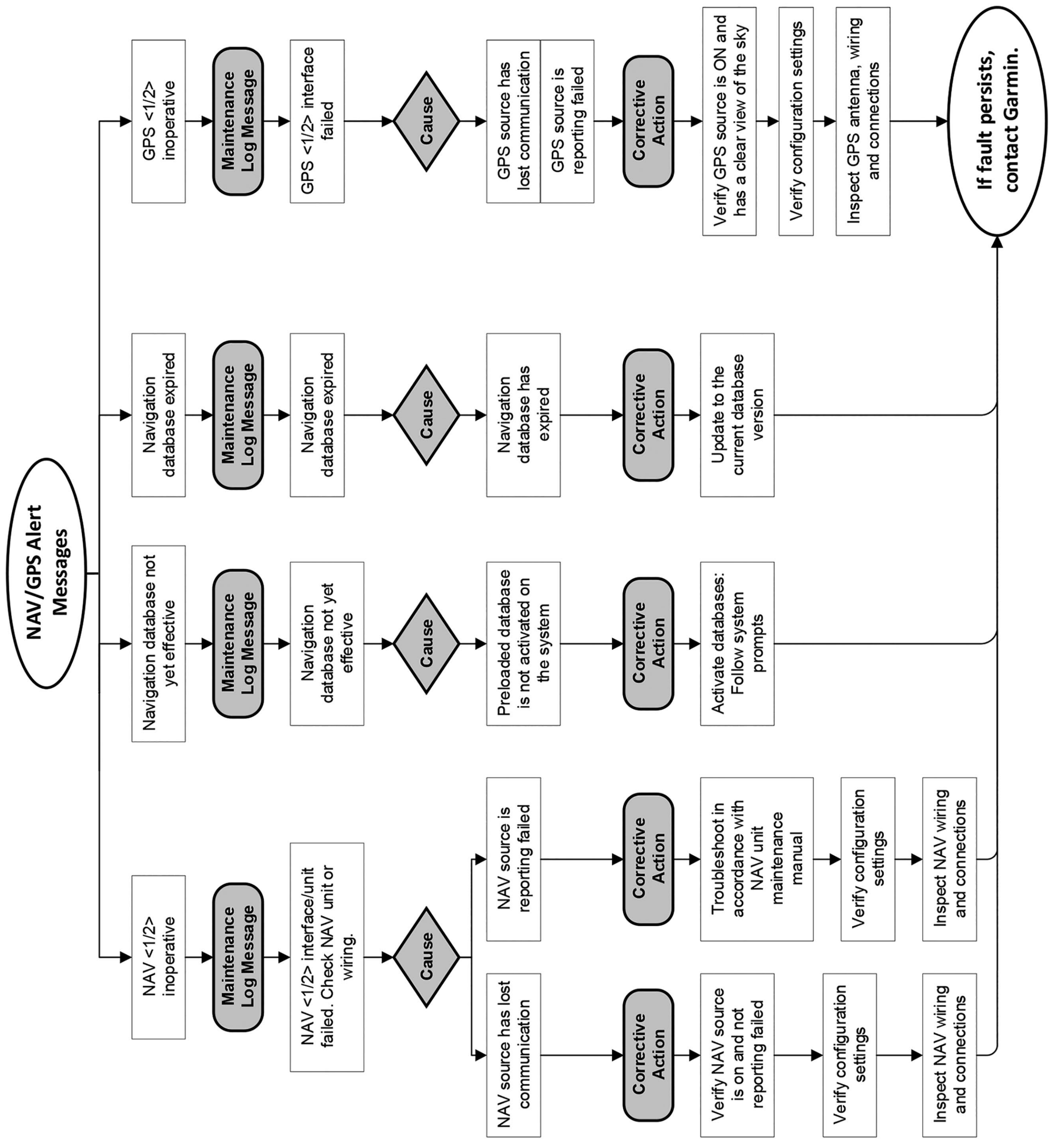
GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 5 of 9)



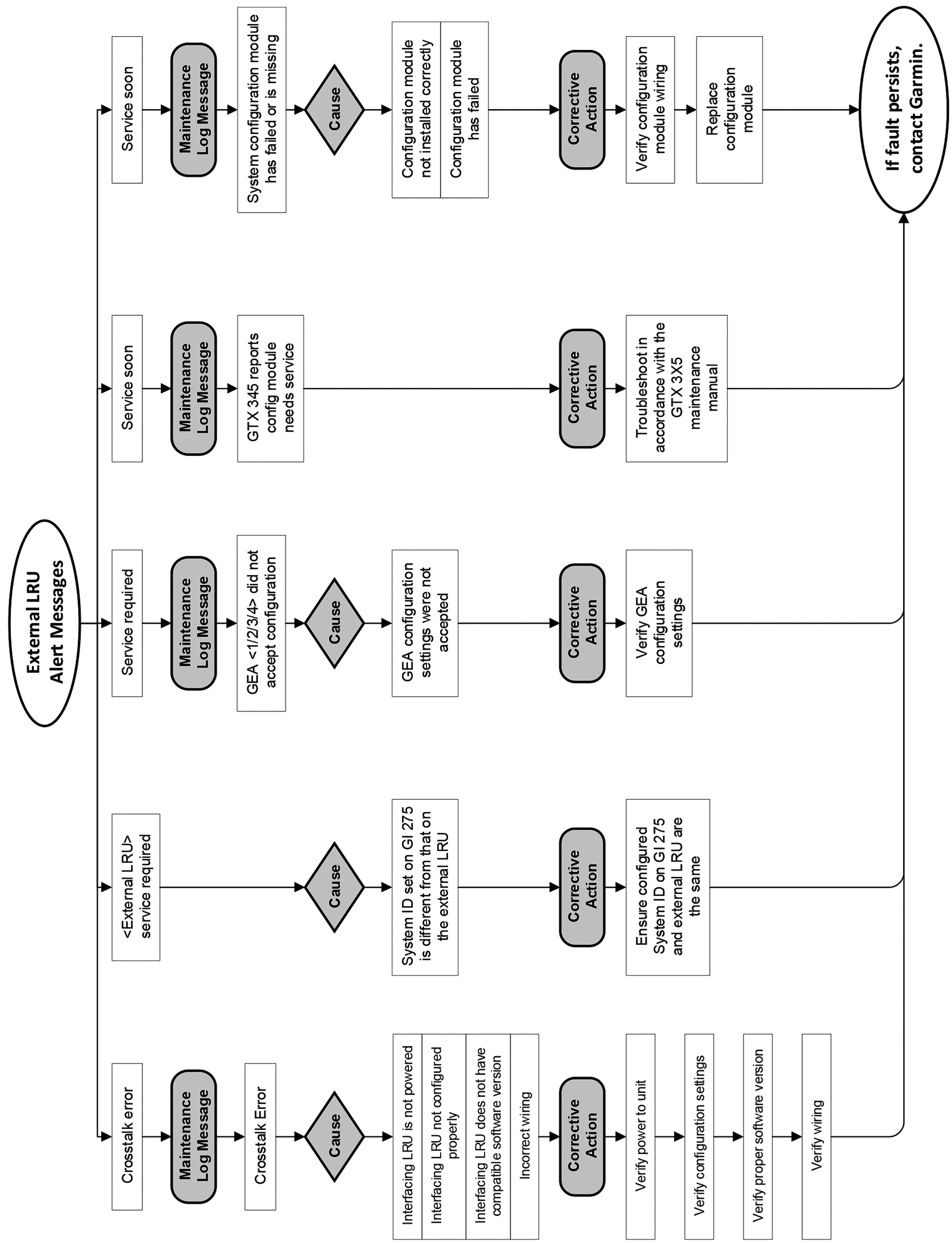
GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 6 of 9)



GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 7 of 9)



GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 8 of 9)



GI 275 Troubleshooting Flowcharts
Figure 2 (Sheet 9 of 9)

3. Inspections

A. Annually (Each 12 Calendar Months)

(1) Conduct a visual inspection of the GI 275 system, LRU's, and associated wiring harnesses to ensure installation integrity per "Inspection" on page 34240114.

(2) Perform a battery capacity check of the Garmin GI 275 Standby Instrument battery per "Battery Capacity Check" on page 34240116.

B. Each Twenty-Four (24) Calendar Months

The GI 275 standby altitude displays must be verified per Title 14 of the CFR 91.411 and Part 43 Appendix E, with the following exception to 14 CFR Part 43 Appendix E, paragraph (b)(1):

The tests of sub-paragraphs (iv) (Friction) and (vi) (Barometric Scale Error) are not applicable to the GI 275 due to the internal ADAHRS interface and instrument display being digital.

C. Each Five (5) Years

The GI 275 Integrated ADAHRS utilizes an Earth magnetic field model that is updated every 5 years. If the magnetic model is not up-to-date, the unit will issue an alert upon start-up indicating the model has expired. A Service Bulletin containing the updated magnetic field model and instructions for installation can be obtained from the Dealer Resource Center or by contacting Garmin.

D. On Condition

(1) When dirty, the touchscreen display of the GI 275 can be cleaned with a soft cotton cloth dampened with clean water. DO NOT use any chemical cleaning agents. Care should be taken to avoid scratching the surface of the display.

(2) Over time, the backlight lamp may dim and the display may not perform as well in direct sunlight conditions. The user must determine by observation when the display brightness is not suitable for its intended use. Contact a Garmin authorized repair station when the backlight lamp requires service.

4. Components

A. GI 275

See "Figure 3" on page 34240115.

The Garmin GI 275 is a multi-functional touchscreen instrument that is installed as a standby instrument displaying Attitude, Altitude, Airspeed, OAT, Heading, and fits in a standard 3-1/8-inch instrument cutout. The semi-round display accepts user input through the use of the capacitive touchscreen display and dual inner and out rotatory knobs on the bottom left of the instrument. The GI 275 is powered by the 7.5 amp standby instrument circuit breaker and has an internal lithium-Ion battery used to supply emergency power to the instrument in the event of an electrical failure for up to 60 minutes.

The GI 275 contains integrated air data, altitude and heading sensors to provide a secondary display of information. It is connected to the pitot-static system and additional equipment is installed to support the display including an external gps antenna and external magnetometer.

It is face-mounted to the instrument panel on the far-left side.

(1) Inspection

Operation of the GI 275 system is not permitted unless a visual inspection has been completed every twelve (12) calendar months. Conduct the following visual inspection of the GI 275 system LRUs and associated wiring harnesses to ensure installation integrity:

(a) Inspect all units for security of attachment, including visual inspection of brackets and other supporting structure attaching all units to the airframe.

(b) Inspect all switches, annunciators, knobs, and buttons for legibility.

(c) Visually inspect each unit’s wiring (including electrical bonding straps), overbraid, and connectors for chafing, deterioration, damage, or wear.

(d) Visually check for any signs of corrosion.

(2) Removal

(a) Ensure the current configuration is saved to a USB drive in accordance with the instructions in "Configuration File and Log" on page 3424012.

(b) Set the battery/master switch to the OFF position and ensure no external power is connected.

(c) Remove the pilot's primary flight displays (PFD1) to gain access behind the instrument panel. See 34-25-03.

(d) Remove the three screws securing the GI 275 to the instrument panel.

(e) Disconnect the backshell connectors from the back of the GI 275.

(f) Disconnect the GPS BNC connector.

(g) Disconnect the pitot-static connections.

(h) Remove the GI 275 from the instrument panel.

(3) Installation

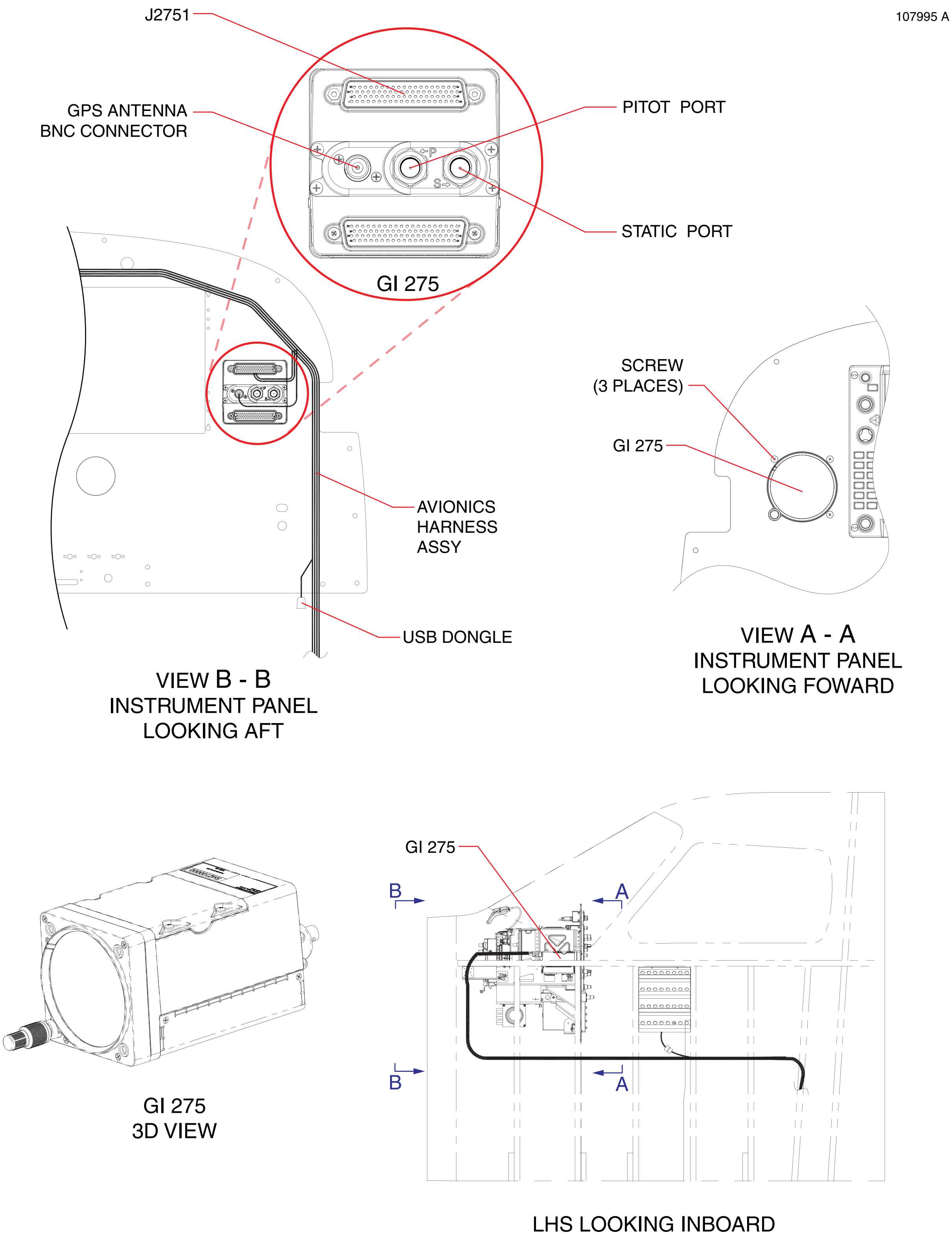
(a) Place the GI 275 into the instrument panel.

(b) Install the three screws to secure the GI 275. Torque screws to 8.0 ± 1.0 in-lbf.

(c) Connect the pitot-static connections.

(d) Connect the wiring connectors and any installed antenna wires to the back of the GI 275 and tighten their retaining screws.

(e) Reinstall the pilot’s primary flight display (PFD1). See 34-25-03.



GI 275 Installation
Figure 3

(4) Return to Service

(a) The System Info page contains the Devices Online page, which reports the status of installed LRUs. The icon next to each LRU reports one of three colored symbols to indicate the status of each LRU. Verify that all LRUs connected or configured to each display have a green checkmark indicator.

(b) The Device Info page (System Info -> Device Info) provides information for each configured LRU as part of the GI 275 system.

1) Touch the Device button and select an interfaced LRU.

2) Verify that all software versions are up-to-date for the interfaced LRU per "Chart 2" on page 34250231.

3) Repeat steps 1) and 2) for each interfaced LRU.

B. Backup Battery

See "Figure 4" on page 34240117.

The GI 275 has an internal lithium-Ion battery used to supply emergency power to the instrument in the event of an electrical failure for up to 60 minutes.

(1) Removal

(a) Remove the GI 275 per "Removal" on page 34240114.

(b) Remove the four screws securing the battery cover to the top of the GI 275 and remove the panel.

(c) Remove the battery by pulling the battery pack straight up until it is unseated from the connector.

(2) Installation

(a) If installing a new battery, remove the electrical contact cover before installation. Retain the battery endcaps.

(b) Ensure proper alignment of the battery contacts and push straight down to seat the connector.

(c) The unit will automatically power on. Power off the unit.

(d) Reinstall the battery access panel and the four panel screws. Torque screws to 8 ± 1 in·lbs.

(e) Reinstall the GI 275 per "Installation" on page 34240114.

(3) Return to Service

(a) The System Info page contains the Devices Online page, which reports the status of installed LRUs. The icon next to each LRU reports one of three colored symbols to indicate the status of each LRU. Verify that all LRUs connected or configured to each display have a green checkmark indicator. This confirms the battery is connected properly.

(b) If the battery was replaced, perform "Battery Capacity Check", below.

(4) Battery Capacity Check

Perform this procedure every twelve (12) calendar months. This procedure will analyze the voltage and discharge qualities of the installed backup battery. The procedure is required to be completed when a battery is installed in the system. A fault indication message will be displayed in Normal mode until this procedure is completed.

NOTE: The Battery Rundown Test may take up to 120 minutes to complete.

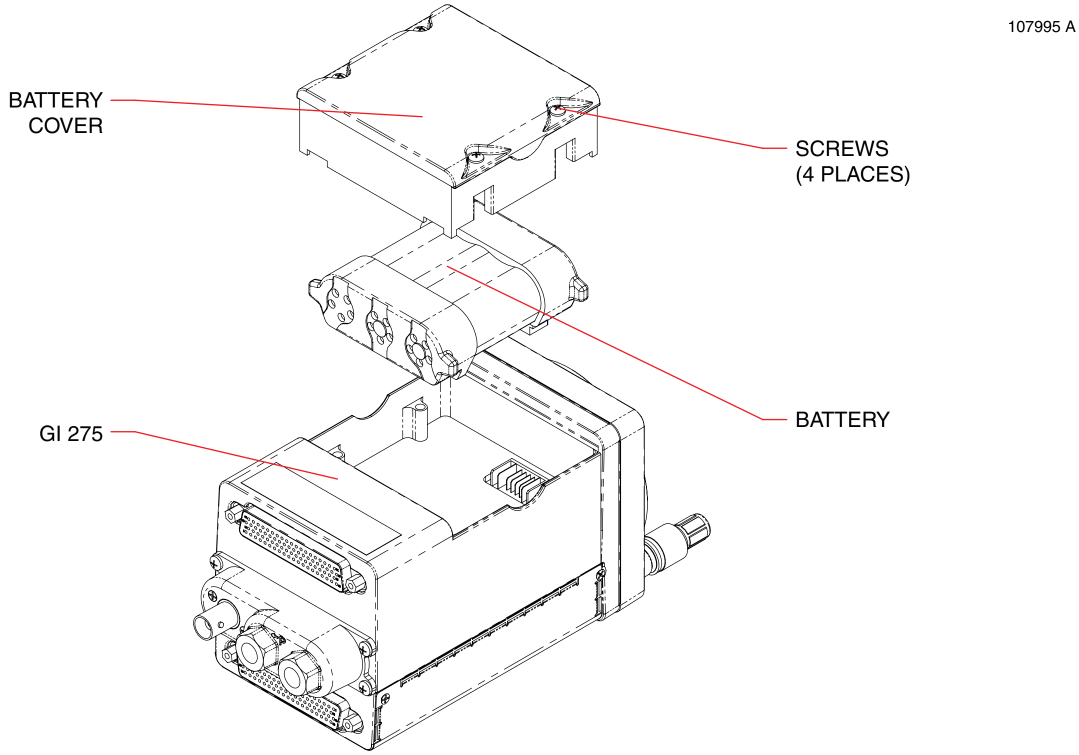
NOTE: The Battery Rundown Test date is reported in UTC.

(a) Power the GI 275 in Configuration mode, (see "Configuration Mode" on page 34240122).

(b) Navigate to the Calibration/Test -> Backup Battery Test page.

(c) Touch the Before Test Checklist button.

(d) Verify that “Discharging” isn’t displayed under Battery State.



GI 275 Backup Battery Installation
Figure 4

(e) Select Test Date and enter the current date.

(f) Complete the on-screen Before Test checklist. Touch each checklist item once completed (if not already checked). Once all checklist items have a green check mark, touch the Back button.

(g) Select Start Test and follow the on-screen commands.

(h) The GI 275 will power off automatically when the test is complete.

(i) Power up the unit in Configuration mode and verify a PASS was achieved by touching the Test Results button and then the Rundown Test Results button (within Calibration/Test › Backup Battery Test.

(5) Battery Storage Maintenance

The GI 275 battery should be kept partially charged when unused for extended periods of time and should not be stored completely discharged. For maximum battery longevity, store within a temperature range of -4°F to 68°F (from -20°C to 20°C). Charge the battery to 30% within 1 year of receipt and recharge to 30% every 2 years thereafter if the GI 275 is not in use.

C. GPS Antenna

See "Figure 5", below, and "Figure 6" on page 34240119.

The Garmin GPS antenna feeds a GPS receiver inside the GI 275 which provides position, velocity, and time data to support various functions. The GPS antenna is installed on a mounting bracket located under a cutout in the glareshield on the co-pilot's side.

(1) Removal

(a) Set the battery/master switch to the OFF position and ensure no external power is connected.

(b) Remove the primary flight displays (PFD's) to gain access behind the instrument panel. See 34-25-03.

(c) Disconnect the GPS antenna's BNC connector from the GI 275.

(d) Remove the two screws, washers, and nuts securing the GPS antenna to the mounting bracket.

(e) Lift up the GPS antenna off the mounting bracket and remove the antenna.

(2) Installation

(a) Place the GPS antenna on top of the mounting bracket.

(b) Secure the antenna to the mounting bracket with the two screws, washers, and nuts. The slotted holes in the bracket allow for adjustment of the antenna during installation.

(c) Route the GPS antenna cable to the GI 275 along the existing avionics harness.

(d) Connect the GPS antenna's BNC connector to the GI 275.

(e) Install the PFD's into the instrument panel. See 34-25-03.

(3) Return to Service

Perform the Backup GPS Signal Check as follows:

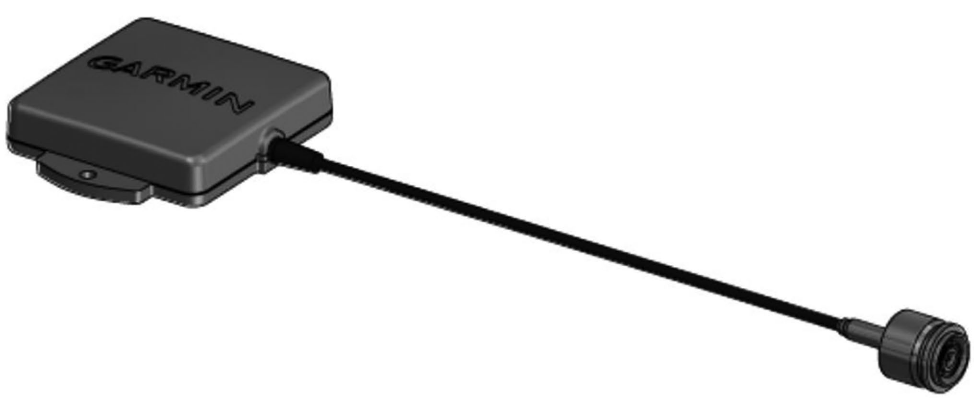
(a) Power on the GI 275 system in Normal mode and verify that the aircraft has an unobstructed view of the sky (or GPS repeater coverage).

(b) Wait at least 5 minutes to allow the VFR GPS to acquire a position.

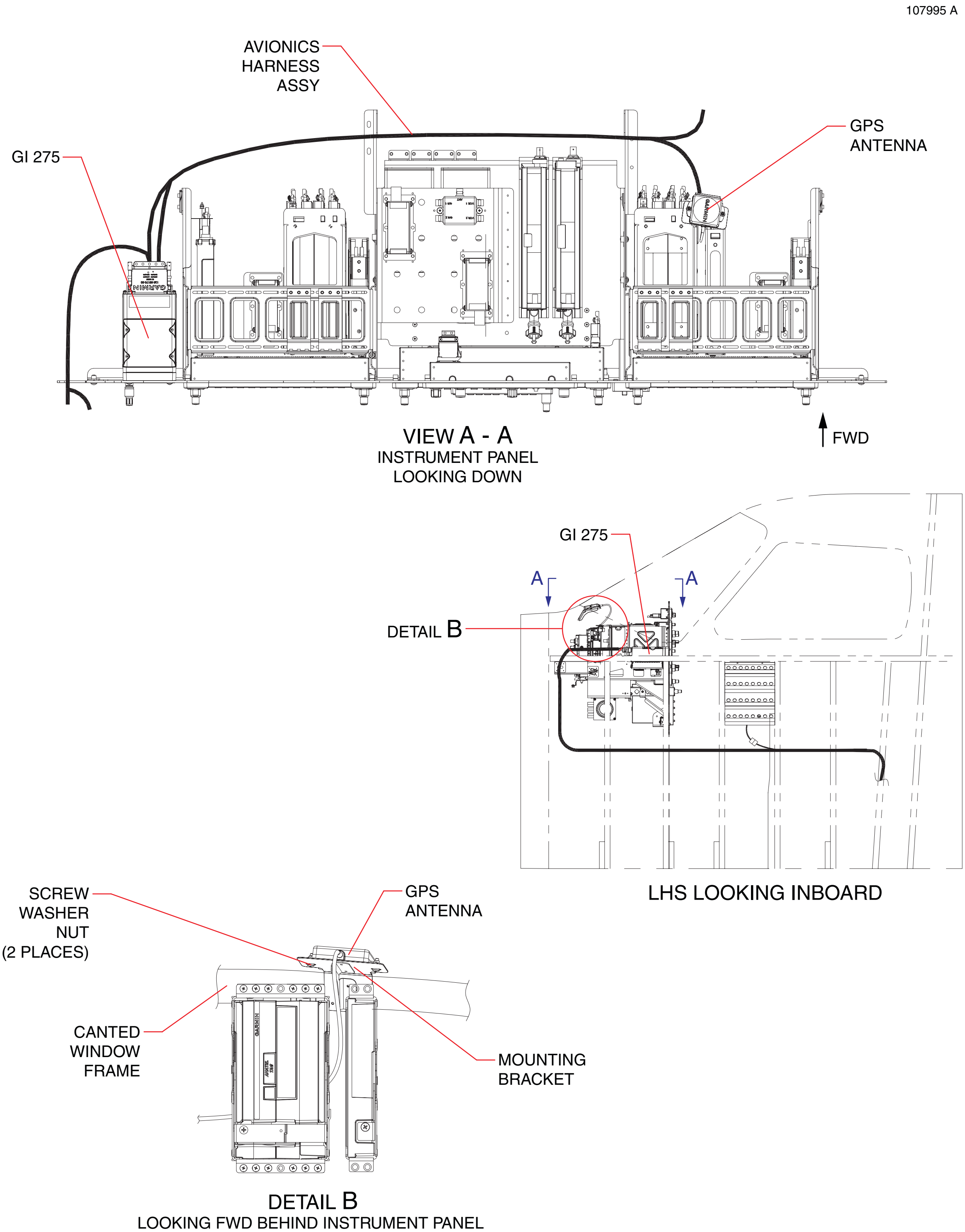
(c) Verify message icon is annunciated on the top-left of the display.

(d) Open the menu and touch Messages (Msgs). Verify that the message "VFR GPS is being used" is present.

(e) Verify ownship symbol is displayed on the map.



Garmin GPS Antenna
Figure 5



Garmin GPS Antenna Installation
Figure 6

D. GMU 11 Magnetometer

See "Figure 7", below, and "Figure 8" on page 34240121.

The GMU 11 Magnetometer is a remote mounted device using magnetic field measurements to provide electronically stabilized heading data to the GI 275. The GMU 11 is installed on an access panel in the bottom skin of the left wing between W.S. 213.23 and 223.22.

(1) Troubleshooting

See "Magnetometer Interference Check" on page 34240128.

(2) Removal

(a) Set the battery/master switch to the OFF position and ensure no external power is connected.

CAUTION: USE NON-MAGNETIC TOOLS ONLY WITHIN 12 INCHES OF THE MAGNETOMETER.

(b) Gain access to the GMU 11 by removing the appropriate access panel. Carefully remove the access panel as the GMU 11 and mounting bracket are installed on the access panel.

(c) Disconnect the GMU 11 connector.

(d) Remove the four (4) screws, washers, and nuts securing the GMU 11 to the mounting bracket.

(3) Installation

CAUTION: USE NON-MAGNETIC TOOLS ONLY WITHIN 12 INCHES OF THE MAGNETOMETER.

(a) Secure the GMU 11 to the mounting bracket using the four (4) screws, washers, and nuts. Torque to 8.0 ± 1.0 in·lbs.

(b) Connect the GMU 11 connector.

(c) Reinstall the access plate taking care to ensure the arrow on the mounting bracket is pointing forward.

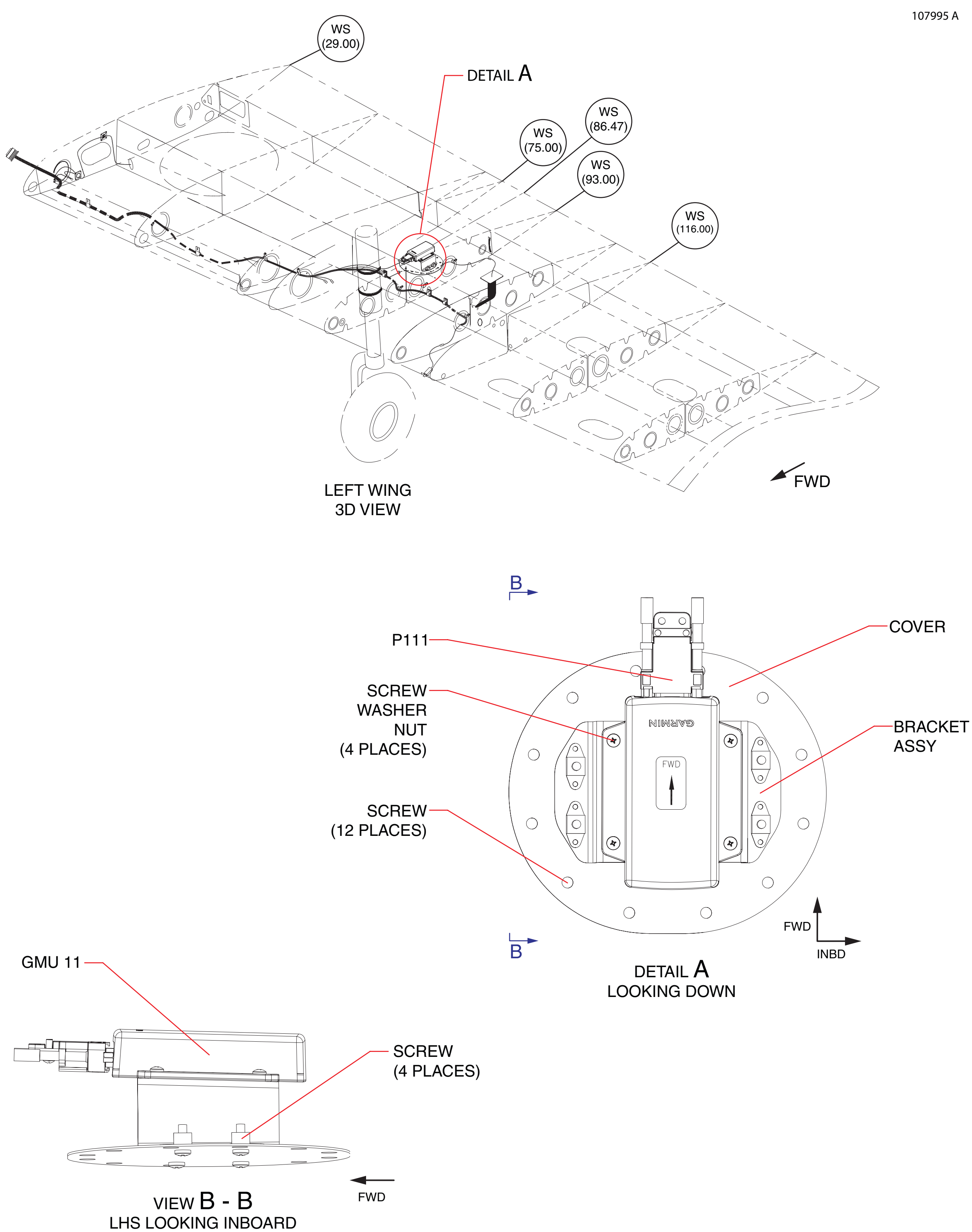
(4) Return to Service

Removal, re-installation, or replacement of the GMU 11 requires a recalibration of the AHRS. Additionally, any removal or addition of electrical components or ferrous materials within 10 feet of the GMU 11 also will require recalibration of the AHRS. Proceed as follows:

(a) Perform "GMU 11 Magnetometer Calibration" on page 34240127.

(b) Perform Vibration Test under "Engine Run Procedures" on page 34240126.

GMU 11 Magnetometer
Figure 7



GMU 11 Installation
Figure 8

5. Post Installation Setup and Checkout Procedures

(PIR-46G34A025, Rev. B.)

A. Required Test Equipment

(1) Ground power equipment
(2) Aircraft jacks
(3) Digital Level (6 inch size)
(4) Digital Level (24–36 inch size)

B. System Configuration

(1) Configuration Mode

(a) Apply power to the GI 275 while holding down the inner knob (located at the bottom-left of the unit).
(b) Hold the knob until the screen reads “By choosing accept below you are confirming you are qualified to modify configuration and understand the risks of doing so.” Then release the knob and press Accept.

(2) Unit ID and Type

(a) Power on the GI 275 unit in configuration mode. (See "Configuration Mode", above.)
(b) Press Unit Type, Unit ID, LRU, and select GI 1.
(c) Select Restart to apply the assignment. Display will reboot in config mode.
(d) Press Unit Type, System ID Source. If bar under Master is not green, press Master to set the display as the master for System ID. Select Yes to confirm the change.

(3) Software Loading Procedure

NOTE: If the GI 275 unit reports the same software P/N and version as indicated in "Chart 3" on page 34240130, it is not required to complete this section.

(a) Power on the GI 275 unit in "Configuration Mode" on page 34240122.
(b) Insert a properly formatted USB drive into the USB dongle (refer to "Garmin GI 275 Software Loader Card Creation Procedure" on page 34240130.
(c) Navigate to the Import SW/Config GI-275 Software page.
(d) Select Loader Card.
(e) Select the appropriate SW versions to load.
(f) Select Update Package at the bottom.
(g) Select Begin Update. A restart is required when completed.
(h) Remove usb drive from usb dongle.

(4) Device Configuration

(a) Power on the GI 275 unit as described in "Configuration Mode", above.
(b) Import the configuration as follows, or manually configure per step 3:

1) Insert the USB drive containing the configuration files into the USB dongle.
2) Navigate to Import SW/Config Import Config.
3) Touch the Import Configuration button.
4) Touch the Select Files button and select the configuration file to be imported.
5) Touch the Select Configuration button.
6) Select all configurations and then touch the Back button.
7) Touch the Import Config () button and then touch the Start button.
8) After config import is complete, remove usb drive from dongle.

(c) If step (b) was skipped, manually configure the settings per "Chart 1" on page 34240122.

CHART 1 (Sheet 1 of 3)
GI 275 CONFIGURATION SETTINGS

Config Page	Config Option	Configuration Setting		
Unit Type Unit Cnfg (within Unit Type)	Airframe Type	Fixed Wing		
	Instrument Type	ADI		
	Standby	Disabled		
	Unit Location	Pilot		
Interfaces	GI 275s Installed	GI 1	Enabled	
		All others	Disabled	
	Wireless	Wireless LRU	GI 1	
	ADC 1	Internal		
	ADC 2	None		
	AHRS 1	Interface	Internal	
		Magnetic Hdg	Interface	GMU 11
			RS232	Port 1
			Orientation	Forward
		Inertial-aided Vertical Speed	Disabled	
		External GPS Aiding	Disabled	
	AHRS 2	None		
	GPS 1	None		
	VFR GPS	Internal		
	All Others	None		
Airframe Cnfg (within Setup)	Roll Pointer	Fixed		
	ADI Style	4-in-1		
	Altitude Bug	Enabled		
	VS Min	Disabled		
	VS Max	Disabled		
	VS Range	±2000		
	ALT/VS Units	Feet (Range 500ft)		
	IAS Units	KT (Range 60kt)		
	Airspeed Bug	Disabled		

CHART 1 (Sheet 2 of 3)
GI 275 CONFIGURATION SETTINGS

Config Page	Config Option		Configuration Setting	
Airspeed Cnfg (within Setup -> Airframe Cnfg)	Mode		Basic	
	Configuration	VS0	62	
		VS1	73	
		Vfe	112	
		Vno	251	
		Vne	251	
		GLIDE	105	
		Vr	85	
		Vx	95	
		Vy	122	
		Vle	170	
		Vmca	OFF	
		Vyse	OFF	
		All Others	OFF	
Lighting (within Setup)	Enhanced Lighting		Enabled	
	Source Selection		Display Source	Lighting Bus
			Knob Source	Photocell
	Photocell Configuration		Response Time	2 Sec
	Lighting Bus Configuration		Input Type	28V DC
			Response Time	0 Sec
Display Lighting Day Mode (within Setup -> Lighting Curve Configuration)	Lighting Bus Curve (set these values while Curve is set to Lighting Bus)		Vertex 1	23.5, 0.0
			Vertex 2	55.0, 0.4
			Vertex 3	62.0, 1.1
			Vertex 4	76.0, 4.0
			Max Level	50.00%
			Min Level	0.02%
			Transition	20.00%
	Photocell Backup Curve (set these values after selecting Curve to Photocell Backup)		Vertex 1	1.0, 0.3
			Vertex 2	40.0, 40.0
			Vertex 3	60.0, 60.0
			Vertex 4	80.0, 80.0

CHART 1 (Sheet 3 of 3)
GI 275 CONFIGURATION SETTINGS

Config Page	Config Option	Configuration Setting
Knob Lighting Day Mode (within Setup -> Lighting Curve Configuration)	Vertex 1	1.0, 0.6
	Vertex 2	40.0, 40.0
	Vertex 3	60.0, 60.0
	Vertex 4	80.0, 80.0
	Max Level	100%
	Min Level	0.01%
	Cutoff	80%
Terrain/TAWS (within Setup)	Terrain Mode	Terrain-Proximity
	External TAWS	Not Installed
Miscellaneous	Traffic Color	White
	Altitude Alerter	200 FT Chime
	Database Sync	Pilot Control
	CDI & BARO Side SYNC	Pilot Control
	CDI & BARO Standby SYNC	Pilot Control
	Outside Air Temp	None
Battery (within Setup)	Backup Battery	Enabled
Ownship Icon Config	Icon	Basic Aircraft
	Color	White

C. Pitch/Roll Calibration

(1) Select Calibrate Pitch/Roll (Home › Calibration/Test -> Attitude/Heading -> Calibrate Pitch/Roll).

(2) Complete the steps listed on the display. Press enter, then touch the Start button to begin the calibration procedure.

(3) Follow the on-screen command prompts.

D. Engine Run Procedures

(1) Vibration Test

(a) Start the aircraft engine in accordance with the aircraft POH.

(b) Power on the display in Configuration mode. Select Engine Vibration Test (Home › Calibration/Test › Attitude/Heading).

(c) Complete the steps listed on the display. Touch the Start button to begin the calibration procedure.

(d) Increase power lever to full power and back to idle slowly over one to two minutes.

(e) If the maximum vibration os less than 100% the test is a Pass. A failure is indicated by a maximum over 100% and "Limit Exceed"message. Verify whether test has passed or failed before pressing Done.

NOTE: In some aircraft, attempting the engine run-up test on a day with very strong and/or gusty winds may cause the test to occasionally fail. However, a failure during windy conditions cannot not be taken as evidence that the test would pass in calm conditions; an actual pass is required before the installation can be considered adequate.

(2) GMU 11 Magnetometer Calibration

(a) Start the aircraft engine per the POH/AFM.

(b) Taxi the aircraft to a desired calibration area.

(c) Power on the display in Configuration mode. (See "Configuration Mode" on page 34240122.)

(d) Navigate to the calibration page (Calibration/Test -> Attitude/Heading -> Calibrate Magnetometer).

(e) Complete the Before Calibration steps listed on the display; touch Next after completing each step to move to the next step.

(f) Touch the Start button when it becomes available to start the calibration procedure.

(g) Follow the on-screen commands to complete the calibration.

NOTE: A successful heading calibration point is a countdown followed by instruction to move. Due to the difficulties in executing smooth, accurate turns, the display may incorrectly interpret the approach heading point and instruct to “HOLD POSITION” prior to full completion of a 36° turn. If this condition is encountered, use outside references to complete the approximate 36° turn, instead of using the display instructions of when to complete the turn (use the compass rose radial to make the 36° (±5°) turn increments). Accurately completing each 36° heading point for the required time as instructed will result in a successful calibration.

NOTE: Due to high winds or excessive airframe vibration, the operator may encounter a condition where the countdown is restarted without full completion of the previous countdown. If this is experienced more than once for a given heading point, the operator should begin turning to the next station (approximately 36°). A minimum of two successful heading points per quadrant is required. It may sometimes be required to hold at a station after a countdown restart. A maximum of 20 heading points are allowed for the entire calibration procedure. If too many countdown restarts are encountered, the calibration will fail with the message, “TOO MANY STATIONS”.

E. Post Installation Checks

(1) Backup Battery Test

Perform "Battery Capacity Check" on page 34240116.

(2) Initial Power On Checks

(a) Apply power to the aircraft and aircraft systems and allow the aircraft systems to initialize.

(b) On the GI 275 Attitude display, verify the following;

- No red-X's displayed
- Attitude display
- Heading display
- Airspeed Tape (Digital read out will be dashes)
- Altitude Tape (Digital readout should be reasonable to field elevation)
- Slip/skid display
- Barometric Pressure setting field

(3) AHRS Sensor Test

Verify that correct aircraft attitude information is present on the Attitude Indicator portion of the EBD (i.e. pitch, roll & slip/skid indicators are all present and reflect current aircraft attitude similar to PFD1 & PFD2).

(4) Magnetometer Interference Check

(a) With the GI 275 in configuration mode (see "Configuration Mode" on page 34240122.), initiate the magnetometer interference test procedure by performing the following steps:

(b) Navigate to the test page (Calibration/Test -> Attitude/Heading -> Interference Test). Press Enter.

(c) Follow the instructions on each page displayed on the GI 275. Press Next.

(d) Select "Start" to begin the test procedure.

(e) Follow the procedure in "Chart 2" on page 34240129, (skipping any items that are not installed in the aircraft).

(f) When the test sequence is completed, wait approximately five seconds and then select "Done" to complete the test.

(g) The GI 275 informs the operator if the installation has passed or failed the magnetometer interference test. The results displayed indicate the worst case percentage of limit and the time at which it occurred. If the test passes, no further action is required.

(h) If the test fails, the installation should be considered unreliable until the source of magnetic interference is identified and remedied. The magnetometer interference test must be repeated until passed. When the magnetometer interference test fails, record the magnetometer maximum deviation value and its corresponding timestamp. Compare the corresponding timestamp with the test sequence to identify which action produced the problem.

CHART 2
MAGNETOMETER INTERFERENCE TEST

ELAPSED TIME Since start of test in seconds	ACTION
0	Test Begins
10	Vent Fan On
20	Vent Fan Off
30	Blower Motor on Low
40	Blower Motor off
50	Blower Motor on High
60	Air Conditioner On
70	Air Conditioner Off
80	Blower Motor Off
90	Nav Lights On
100	Nav Lights Off
110	Surface De-Ice On
120	Fuel Pumps MAN
130	Fuel Pumps OFF
140	Surface De-Ice Off
150	Landing Lights On
160	Landing Lights Off
170	Recognition Lights On
180	Recognition Lights Off
190	Strobe Lights On
200	Strobe Lights Off
210	WX Radar On
220	WX Radar Standby
230	Autopilot Engaged
240	Autopilot Disengaged

CHART 3
REQUIRED EQUIPMENT LIST (REL)

GARMIN GI 275 SOFTWARE CONFIGURATION
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	Remarks
GI 275	006-B3173-03	2.20	
GI 275 Expansion	006-B3178-03	2.20	
GMU 11	006-B2770-01	2.00	
PIR-46G34A012 Rev. A.			

6. Garmin GI 275 Software Loader Card Creation Procedure

   A. To obtain the latest software version, visit dealers.garmin.com and navigate to the Support / Software Downloads page.

   B. Search by Keyword for GI 275 and download the appropriate self-extracting installation file.

   C. Insert a FAT32-formatted USB drive into USB slot of PC.

   D. Open the self-extracting update file and follow the instructions provided by the installer application. You will be prompted to specify the drive letter of the USB drive you wish to use for the update. When the transfer is complete, safely eject the USB drive.

7. Garmin GI 275 Configuration USB Drive Creation Procedure

   NOTE: This procedure is used to create a configuration USB drive, which can be used to export configuration settings from one GI 275 to be imported on another GI 275. Before performing this procedure to create a USB drive, ensure all settings are correct per "Device Configuration" on page 34240122.

   (1) Insert a blank FAT32-formatted USB drive into the USB dongle.
   (2) Start the GI 275 in configuration mode (refer to section 4.1).
   (3) Navigate to SW/Config -> Config Options page.
   (4) Press the Export Configuration button.
   (5) Touch the Select Name button and enter a name (46G34A025-002 for M600 configuration, 46G34A025-003 for M500 configuration, 46G34A025-004 for M350 configuration).
   (6) Press the Export Config Button. Files will be saved to the USB drive.
   (7) Label the USB drive with the appropriate dash number (46G34A025-002, 46G34A025-003, or 46G34A025-004) and drawing revision.

INTEGRATED AVIONICS SYSTEM (IAS) - GARMIN G1000

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Integrated Avionics System (IAS) - Garmin G1000

(PIR-190-00907-00, Rev B.)

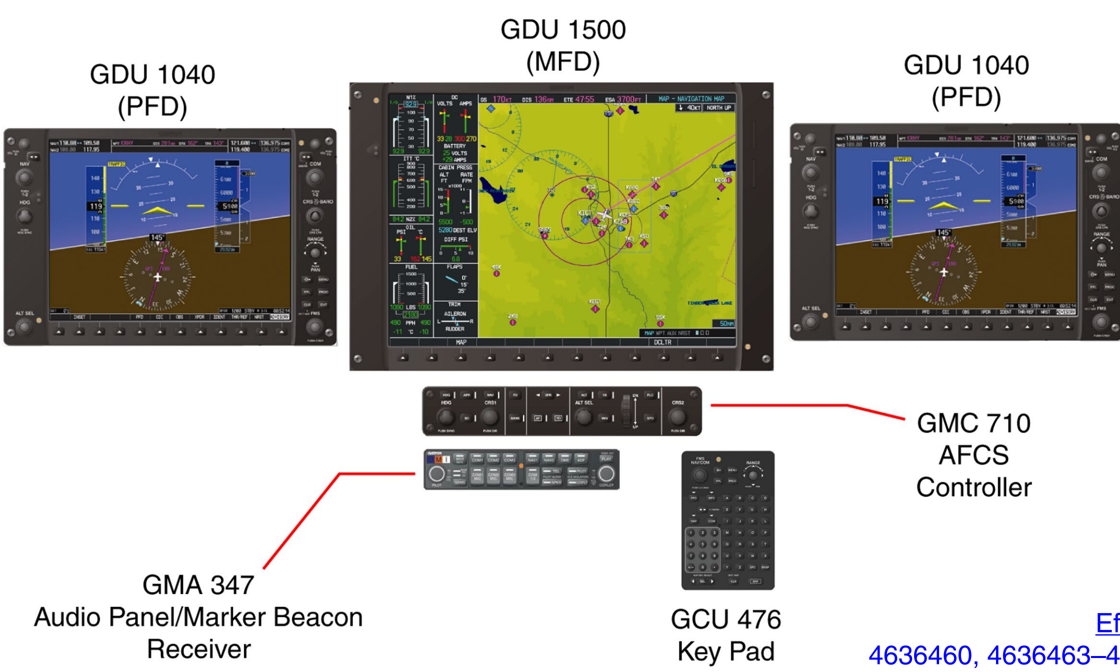
Standard equipment in S/N's 4636460, 4636463–4636715, and 4636717–4636719.

A. Description

See "Figure 1".

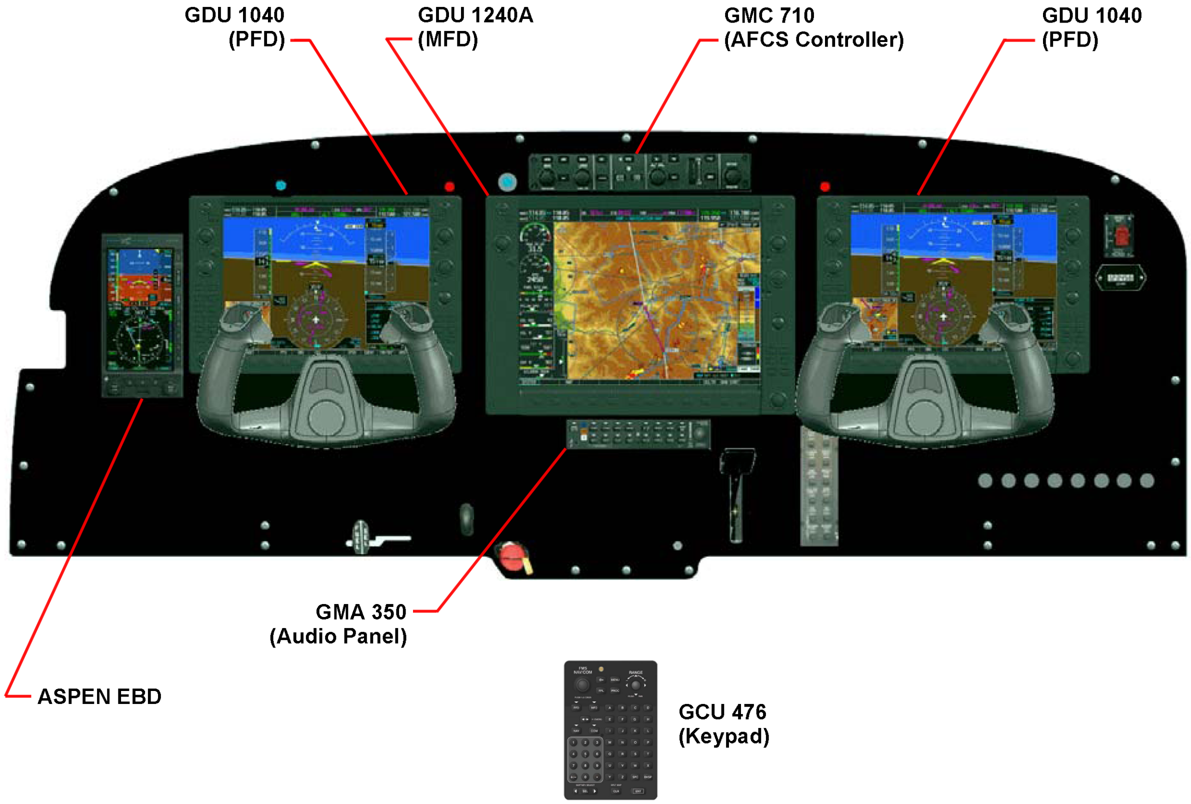
The Garmin G1000 Integrated Avionics System consists of three (3) displays, two (2) GDU 1040s dedicated as primary flight displays (PFD) for the pilot and copilot and a GDU 1500 or GDU 1240A multi-function display (MFD) installed in the center. Functions provided by the system include display of attitude, heading, navigation, traffic air data, engine and airframe status, and a moving map display with position derived by GPS. In addition to display functions, GPS navigation, VHF/Com, VOR/ILS and transponder functions are provided by the G1000 system. Also the Garmin G1000 system integrates the GFC 700 Automated Flight Control System and provides the flight director (FD), autopilot (AP), and yaw damper (YD) functions.

See the latest revision of the Garmin G1000 Pilots Guide for Piper PA-46 Mirage/Matrix (or PA-46 Mirage, as appropriate) for control and operation information. See Vendor Publications under Supplementary Publications in the Introduction.



Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153

Garmin G1000 Display and Controls
Figure 1 (Sheet 1 of 2)



Effectivity
4636663, 4636652 and up

Garmin G1000 Display and Controls
Figure 1 (Sheet 2 of 2)

(1) Flight Instrumentation

See "Figure 2" on page 3425014.

The GRS 77 AHRSs, GDC 74A Air Data Computers, and GMU 44 Magnetometers are responsible for providing the G1000 system with aircraft attitude, heading, altitude, airspeed, vertical speed, and outside air temperature information, all displayed on the PFD (data is displayed on the MFD in reversionary mode only).

Primary data outputs from the GRS and GDC are sent directly to the PFD via ARINC 429. Secondary data paths connect the GRS and GDC to the MFD. Additional communications paths connect the GRS and GDC to both GIA 63W units.

The GRS 77 receives GPS data from both GIAs, airspeed data from the GDC, and magnetic heading from the GMU. Using these three external sources, combined with internal sensor data, the GRS accurately calculates aircraft attitude and heading.

(2) Engine Indicator System

See "Figure 2" on page 3425014.

The GEA 71 provides engine/airframe data to the G1000 system. Analog data is received from various sensors and transducers and is converted to a digital signal by the GEA 71. Digital information is then sent through the primary RS-485 serial path to the #1 GIA 63W. From the GIA, data is sent through the HSDB (ethernet) connection to the PFD, then on to the MFD for display. A backup data path from the GEA to the #2 GIA 63W, then on to the MFD, exists in the event the primary path fails.

(3) Communications/Navigation Systems

See "Figure 2" on page 3425014.

The GIA 63W IAUs contain VHF COM, VHF NAV, and GPS receivers. COM and NAV audio is sent via digital audio to the GMA 347/GMA 350 Audio Panel.

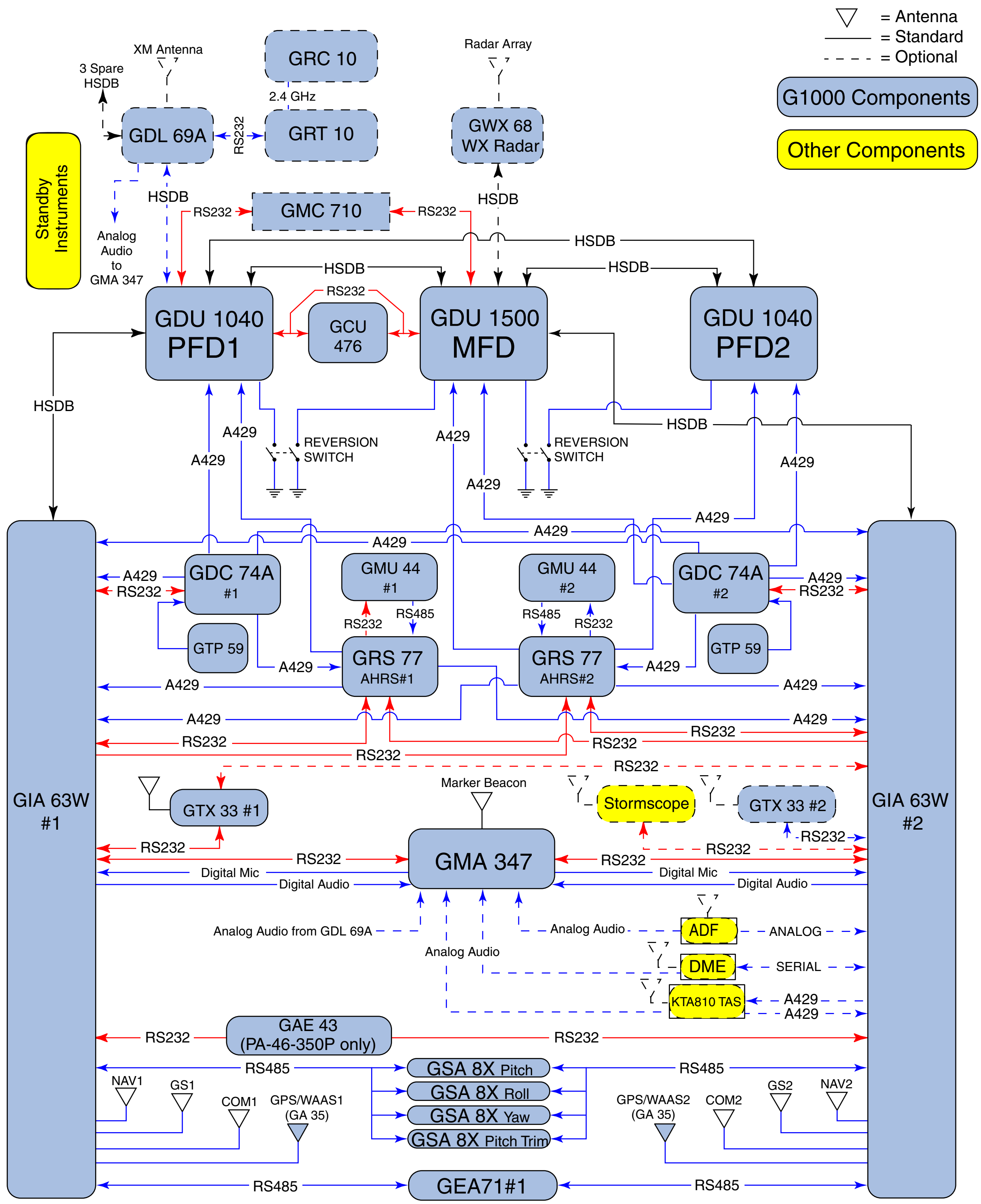
GPS information is sent to the GRS 77 AHRS and to both displays for processing.

The GTX 33 Series Mode S Transponder communicates with both GIAs in a single transponder installation. In a dual installation, each transponder communicates with its on-side GIA. Transponder data is sent from the GIAs to the PFD where control and operation occurs.

(4) Components Details

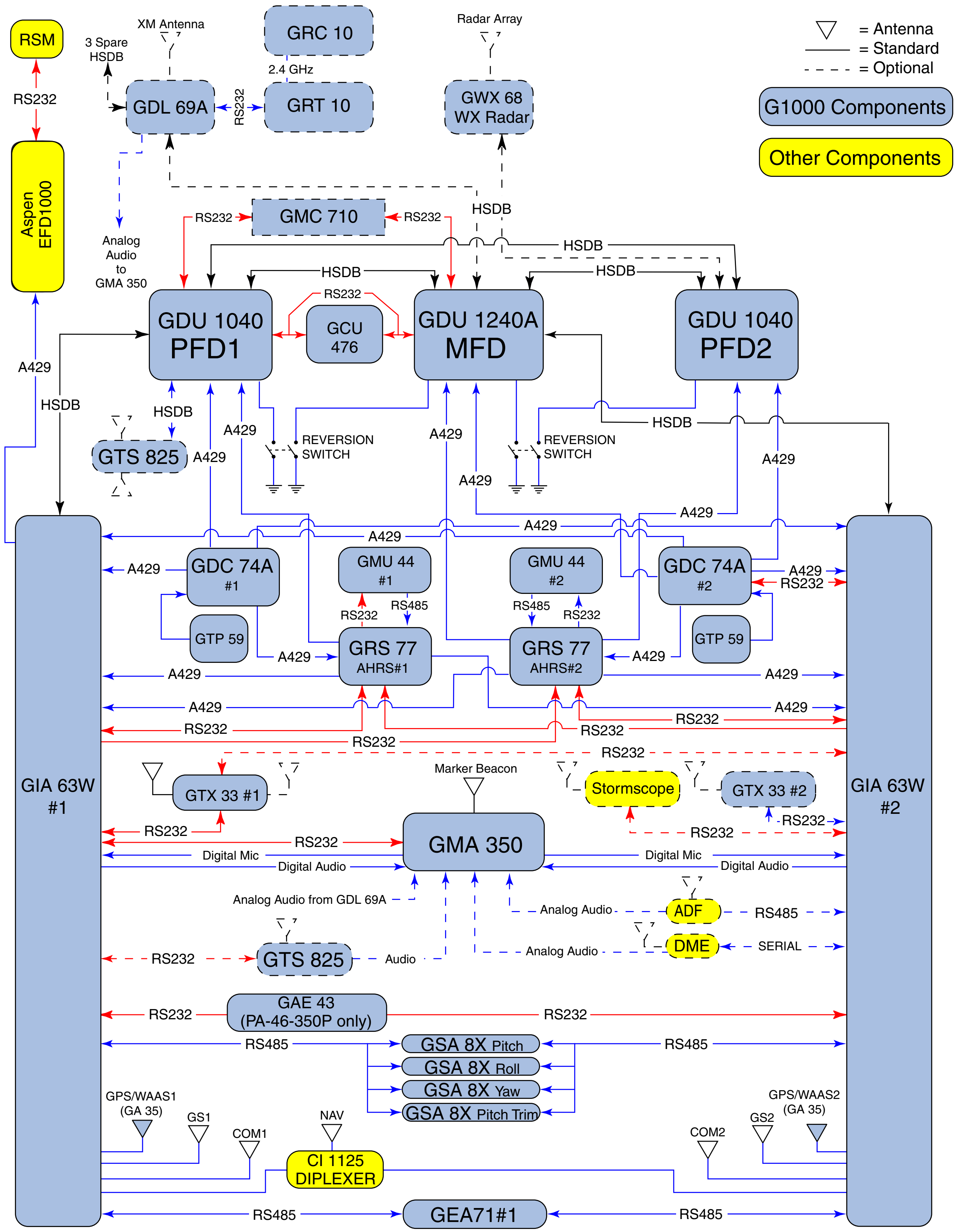
See "Components" on page 34250162, for detailed descriptions of the individual G1000 System Components / Line Replaceable Units (LRU's).

See 22-10-00 for components of the GFC 700 Autopilot System.



Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153

Component Block Diagram
Figure 2 (Sheet 1 of 2)



Component Block Diagram
Figure 2 (Sheet 2 of 2)

Effectivity
4636633, 4636652 and up

B. Power Distribution

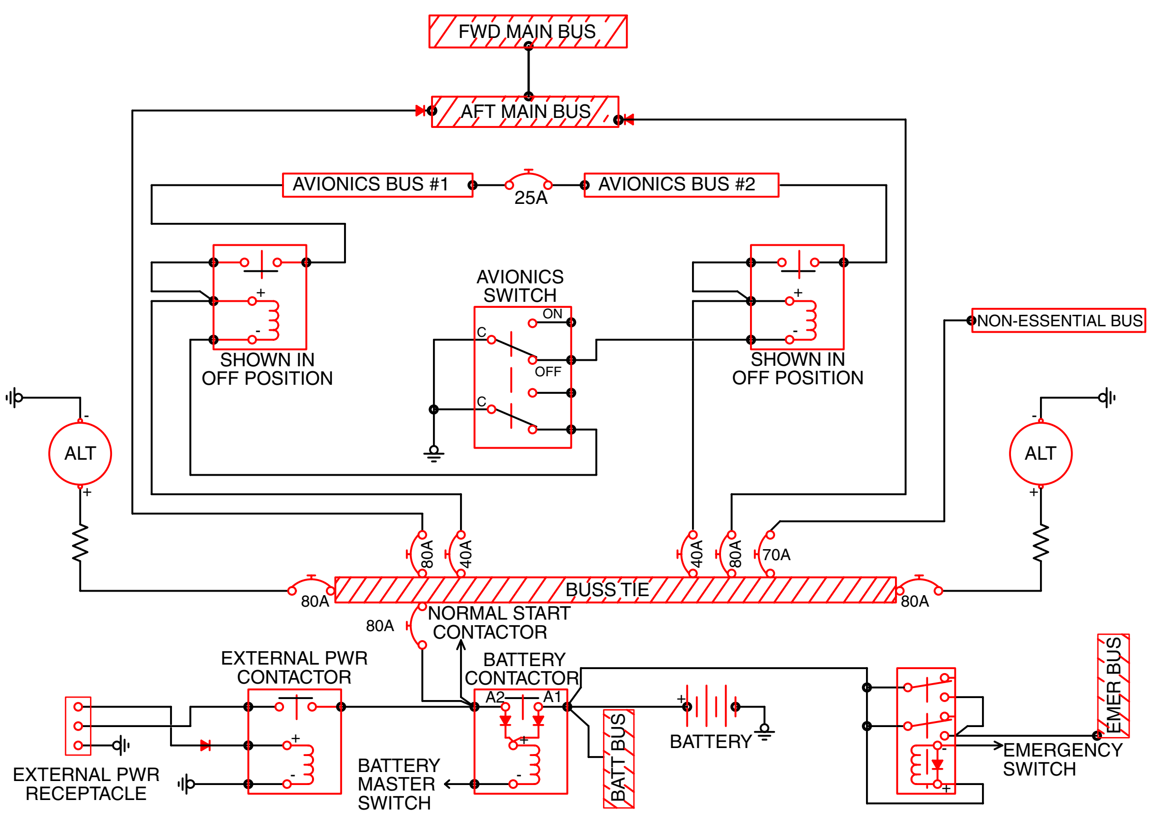
See "Figure 3" and "Figure 4".

Distribution of power to the G1000 system occurs on four aircraft electrical busses.

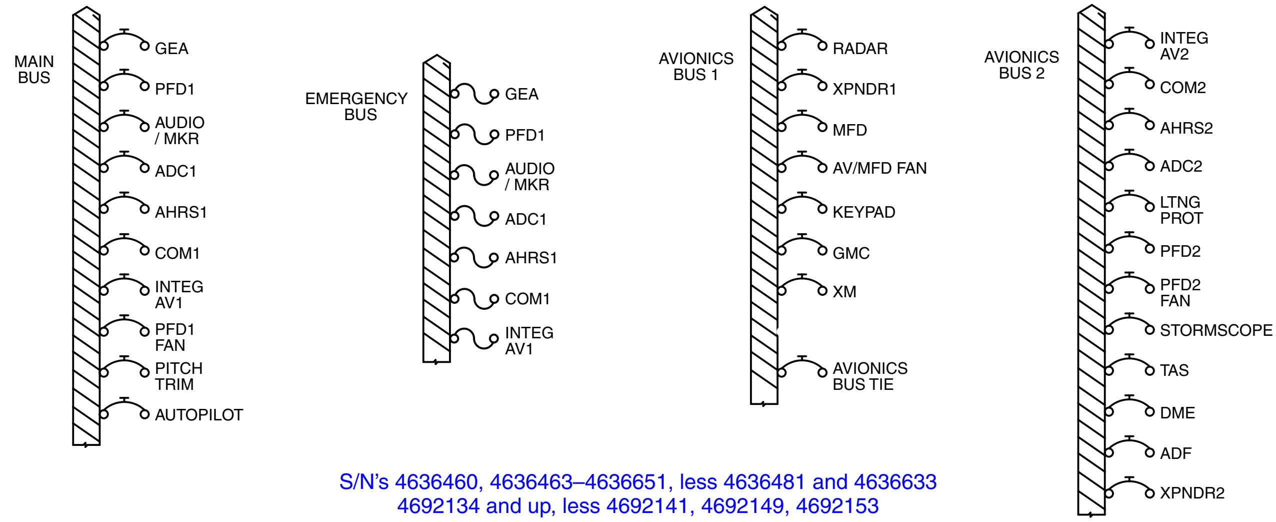
Main Bus: The current is fed to the main electrical bus by two conductors. Two in-line diodes provide isolation in the event of a ground fault in one of the feeder lines. The two feeders are protected by two 80 amp circuit breakers. The non-essential bus is fed by the 70 amp circuit breaker.

Emergency Bus: The Emergency bus exists as a backup power bus to the G1000 system. When the EMER switch is turned on, power must be removed from the Main bus by turning the battery master switch to the “OFF” position. At the same time, the relay connecting the Emergency bus to the emergency battery is closed, providing emergency power to the connected equipment.

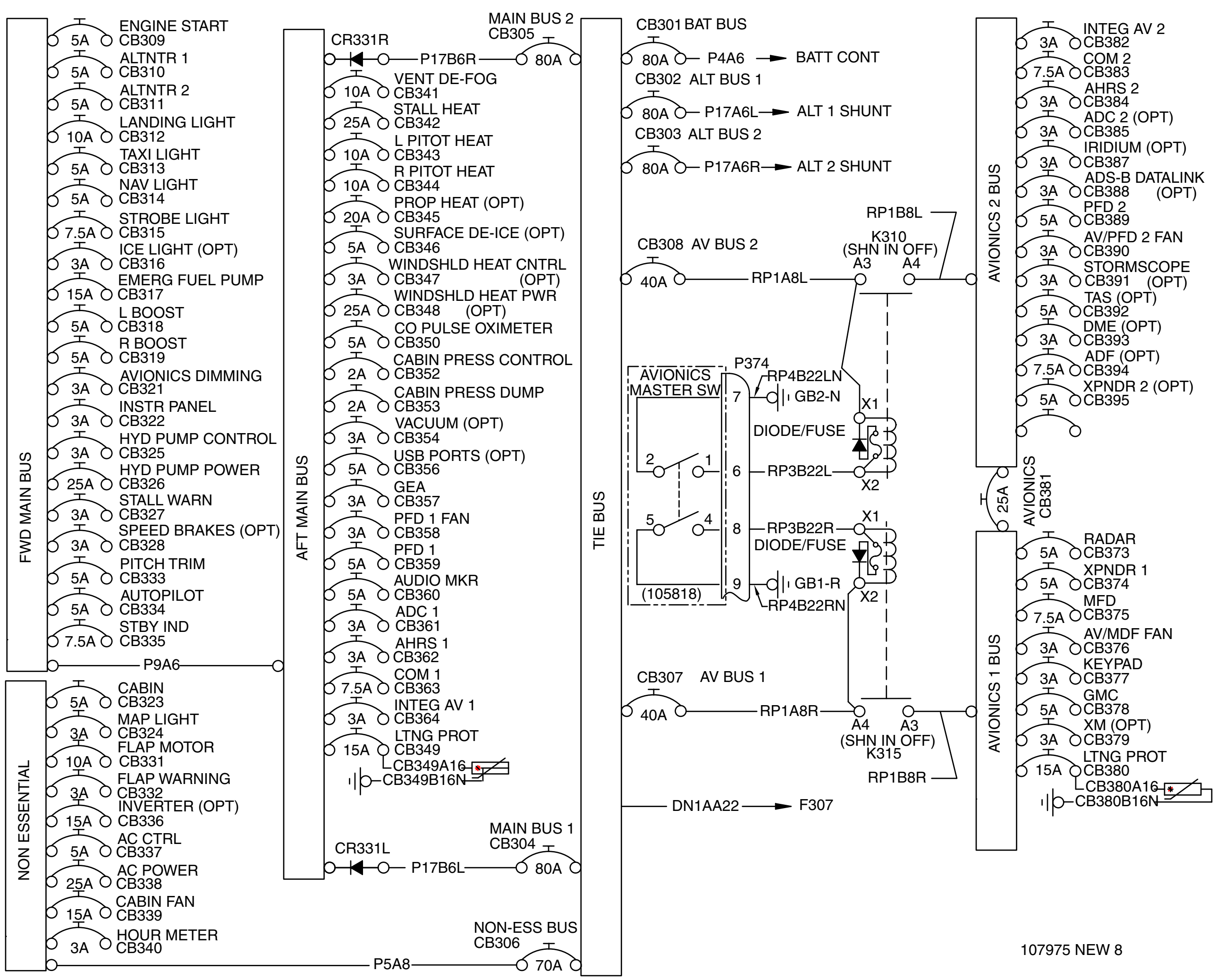
Avionics Bus: The two avionics busses are fed through independent contactors. The feeders to the contactors are protected by 40-amp circuit breakers. The busses are interconnected by a 25-amp bus tie and controlled by the avionics bus master switch. When the AV BUS MASTR switch on the overhead switch panel is pressed, both avionics contactors (normally closed) relax allowing current to flow to both avionics busses. Should the need arise, either avionics bus can be isolated by pulling the avionics bus BUS TIE circuit breaker and the appropriate avionics circuit breaker.



Bus Feeds
Figure 3



S/N's 4636633, 4636652 and up



Component Power Sources
Figure 4

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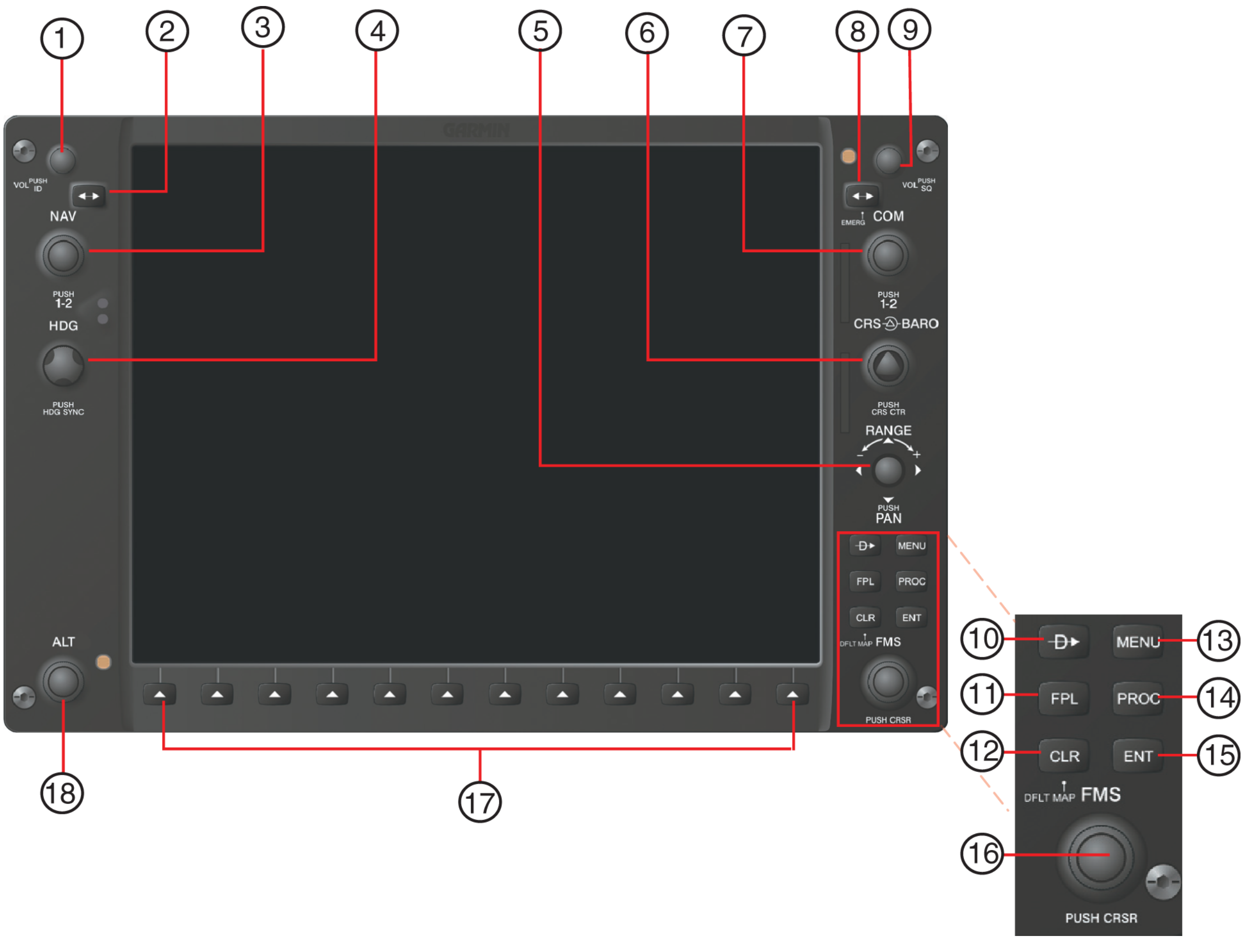
C. G1000 System Controls and Operation

(1) System Controls

The G1000 system controls are located on the PFD bezels, MFD/PFD Keypad Control Unit, AFCS Control Unit and above the respective PFD displays. See "Figure 5" on page 34250110 thru "Figure 11" on page 34250117 for a brief description.

The MFD/PFD Keypad Control Unit is a pedestal-mounted user interface allowing for ease of data entry, MFD/PFD operation, and NAV/COM tuning. Many procedures in this manual can be performed using the MFD/PFD Keypad Control Unit rather than the display bezel controls. Indicators above the PFD, MFD, NAV, and COM keys are illuminated when their respective control mode(s) are selected. The unit is in MFD control mode by default on system power-up. NAV/COM radio tuning can be accomplished in either PFD or MFD control mode. The appropriate frequency box on the selected display is outlined by a light blue selection box, which flashes for a few seconds to indicate Control Unit activity. Selection of a different display control or radio tuning mode results in cancelation of the previous radio tuning mode.

See the appropriate Garmin G1000 Pilot's Guide for Piper PA-46 Mirage/Matirx for detailed information on the operation and controls of the G1000 system and its various components.



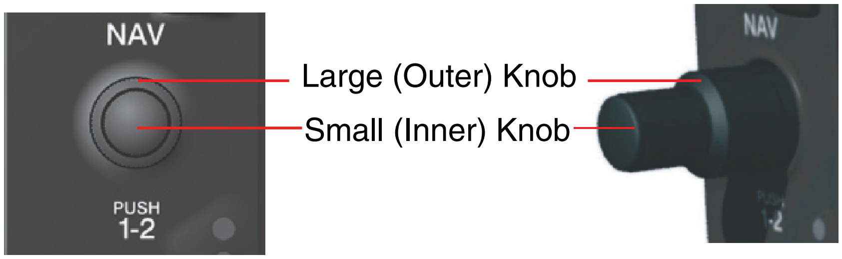
PFD CONTROLS

1 NAV VOL/ID Knob	Turn to control NAV audio volume (shown in the NAV Frequency Box as a percentage) Press to toggle Morse code identifier audio ON/OFF.
2 NAV Frequency Transfer Key	Transfers the standby and active NAV frequencies.
3 NAV Knob	Turn to tune NAV receiver standby frequencies (large knob for MHz; small for kHz) Press to toggle light blue tuning box between NAV1 and NAV2.
4 Heading Knob	Turn to manually select a heading Press to display a digital heading momentarily to the left of the HSI and synchronize the Selected Heading to the and current heading.
5 Joystick	Turn to change map range Press to activate Map Pointer for map panning.

PFD Controls
Figure 5 (Sheet 1 of 2)

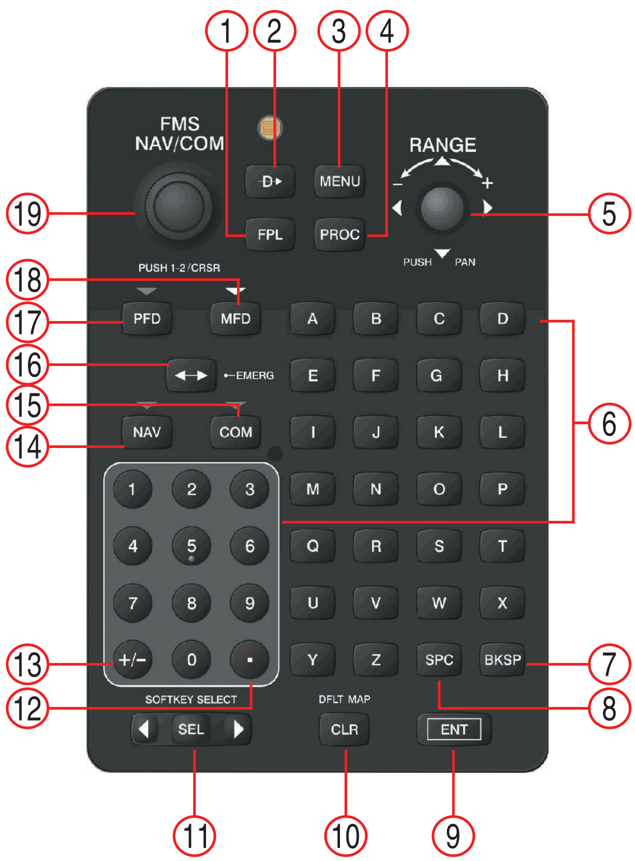
6 **CRS/BARO Knob**	Turn large knob for altimeter barometric pressure setting Turn small knob to adjust course (only when HSI is in VOR or OBS Mode) Press to re-center the CDI and return course pointer directly TO bearing of active waypoint/station.
7 **COM Knob**	Turn to tune COM transceiver standby frequencies (large knob for MHz; small for kHz) Press to toggle light blue tuning box between COM1 and COM2 The selected COM (green) is controlled with the COM MIC Key (Audio Panel).
8 **COM Frequency Transfer Key**	(EMERG) Transfers the standby and active COM frequencies Press and hold 2 seconds to tune the emergency frequency (121.5 MHz) automatically into the active frequency field.
9 **COM VOL/SQ Knob**	Turn to control COM audio volume level (shown as a percentage in the COM Frequency Box) Press to turn the COM automatic squelch ON/OFF.
10 **Direct-to Key** ( )	Activates the direct-to function and allows the user to enter a destination waypoint and establish a direct course to the selected destination (specified by identifier, chosen from the active route).
11 **FPL Key**	Displays flight plan information.
12 **CLR Key**	(DFLT MAP) Erases information, cancels entries, or removes menus Press and hold to display the MFD Navigation Map Page (MFD only).
13 **MENU Key**	Displays a context-sensitive list of options for accessing additional features or making setting changes.
14 **PROC Key**	Gives access to IFR departure procedures (DPs), arrival procedures (STARs), and approach procedures (IAPs) for a flight plan or selected airport.
15 **ENT Key**	Validates/confirms menu selection or data entry.
16 **FMS Knob** (Flight Management System Knob)	Press to turn the selection cursor ON/OFF. Data Entry: With cursor ON, turn to enter data in the highlighted field (large knob moves cursor location; small knob selects character for highlighted cursor location) Scrolling: When a list of information is too long for the window/box, a scroll bar appears, indicating more items to view. With cursor ON, turn large knob to scroll through the list. Page Selection: Turn knob on MFD to select the page to view (large knob selects a page group; small knob selects a specific page from the group).
17 **Softkey Selection Keys**	Press to select softkey shown above the bezel key on the PFD/MFD display.
18 **ALT Knob**	Sets the Selected Altitude, shown above the Altimeter (the large knob selects the thousands, the small knob selects the hundreds).

PFD Controls
Figure 5 (Sheet 2 of 2)



The NAV, CRS/BARO, COM, FMS, and ALT knobs are concentric dual knobs, each having small (inner) and large (outer) control portion. When a portion of the knob is not specified in the text, either may be used.

Concentric Dual Knobs
Figure 6

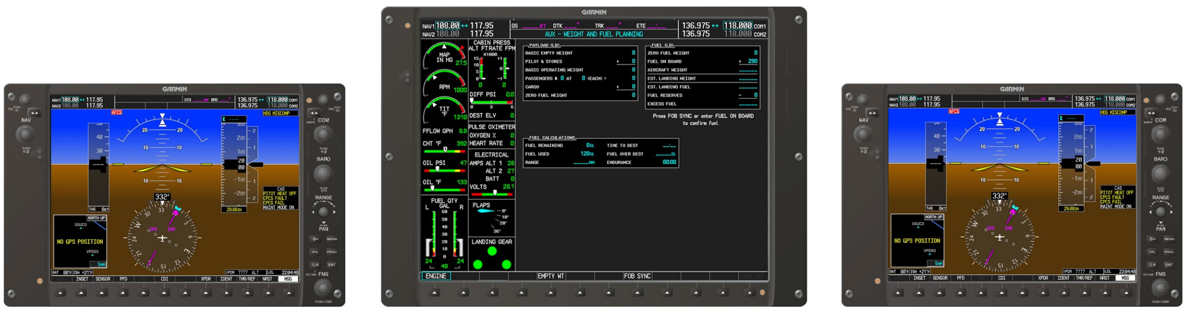


GCU 476 MFD/PFD Keypad Control Unit
Figure 7 (Sheet 1 of 2)

1 **FPL Key**	Displays flight plan information.
2 **Direct-to Key** ( )	Activates the direct-to function and allows the user to enter a destination waypoint and establish a direct course to the selected destination (specified by identifier, chosen from the active route).
3 **MENU Key**	Displays a context-sensitive list of options for accessing additional features or making setting changes.
4 **PROC Key**	Gives access to IFR departure procedures (DPs), arrival procedures (STARs), and approach procedures (IAPs) for a flight plan or selected airport.
5 **Joystick**	Turn to change map range Press to activate Map Pointer for map panning.
6 **Alphanumeric Keys**	Allow data entry (rather than using the FMS Knob to select characters/numbers).
7 **BKSP Key**	Moves cursor back one character space and removes last character entered.
8 **SPC Key**	Adds a space character.
9 **ENT Key**	Validates or confirms a menu selection or data entry.
10 **CLR Key**	Erases information, cancels entries, or removes menus. Press and hold to display the MFD Navigation Map Page (MFD only).
11 **SEL Key**	Arrows move light blue Softkey Selection Box on selected display Press the center to activate the selected softkey.
12 **Decimal Key**	Enters a decimal point character.
13 **Plus-Minus (±) Key**	Toggles entry between the + and - characters.
14 **NAV Key**	Selects/deselects NAV radio tuning mode on the MFD/PFD Control Unit.
15 **COM Key**	Selects/deselects COM radio tuning mode on the MFD/PFD Control Unit.
16 **Frequency Transfer Key**	(EMERG) Transfers between active and standby selected COM or NAV tuning frequencies Press and hold 2 seconds to tune the emergency frequency (121.5 MHz) automatically into the active frequency field.
17 **PFD Key**	When selected, the MFD/PFD Control Unit can be used to access PFD functions.
18 **MFD Key**	When selected, the MFD/PFD Control Unit can be used to access MFD functions (default display control mode).
19 **FMS/NAV-COM Knob**	NAV/COM Tuning Modes: Acts as the NAV or COM Knob PFD/MFD Control Modes: Acts as the FMS Knob.

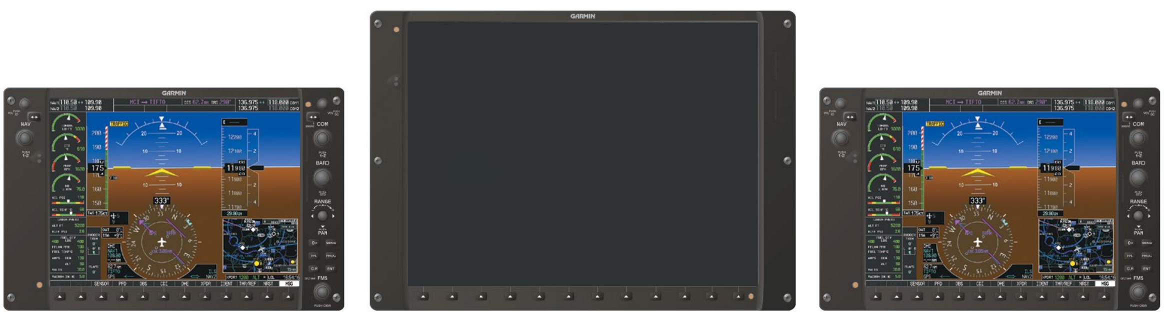
GCU 476 MFD/PFD Keypad Control Unit
Figure 7 (Sheet 2 of 2)

Effectivity: 4636460, 4636463–4636651 less 4636481 and 4636633; 4692134 and up, less 4692141, 4692149 and 4692153

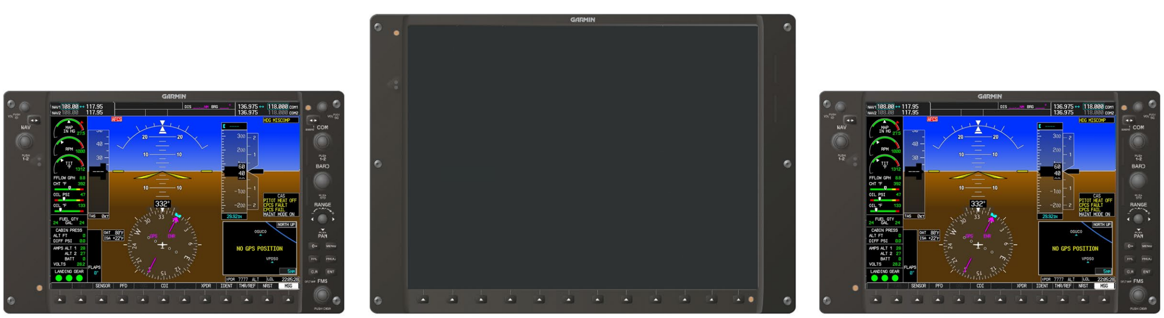


Effectivity: 4636633, 4636652 and up

Normal Mode
Figure 8



Effectivity: 4636460, 4636463–4636651 less 4636481 and 4636633; 4692134 and up, less 4692141, 4692149 and 4692153



Effectivity: 4636633, 4636652 and up

Reversionary Mode MFD Fail
Figure 9

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153

Effectivity
4636633, 4636652 and up

Reversionary Mode #1 PFD Fail
Figure 10

(2) System Operation

The displays are connected together via multiple data busses, thus allowing for high-speed communication. As shown in "Figure 2" on page 3425014, each GIA 63W is connected to the on-side PFD. This section discusses the normal and reversionary modes of operation as well as the various AHRS modes of the G1000 system.

(a) Normal Operation

1) PFD

In normal mode, the PFD presents graphical flight instrumentation (attitude, heading, airspeed, altitude and vertical speed), thereby replacing the traditional flight instrument cluster. The PFD also offers control for COM and NAV frequency selection.

2) MFD

In normal mode, the right portion of the MFD displays a full-color moving map with navigation information, while the left portion of the MFD is dedicated to the Engine Indication System (EIS). "Figure 8" on page 34250114 gives an example of the G1000 displays in normal mode.

(b) Reversionary Mode

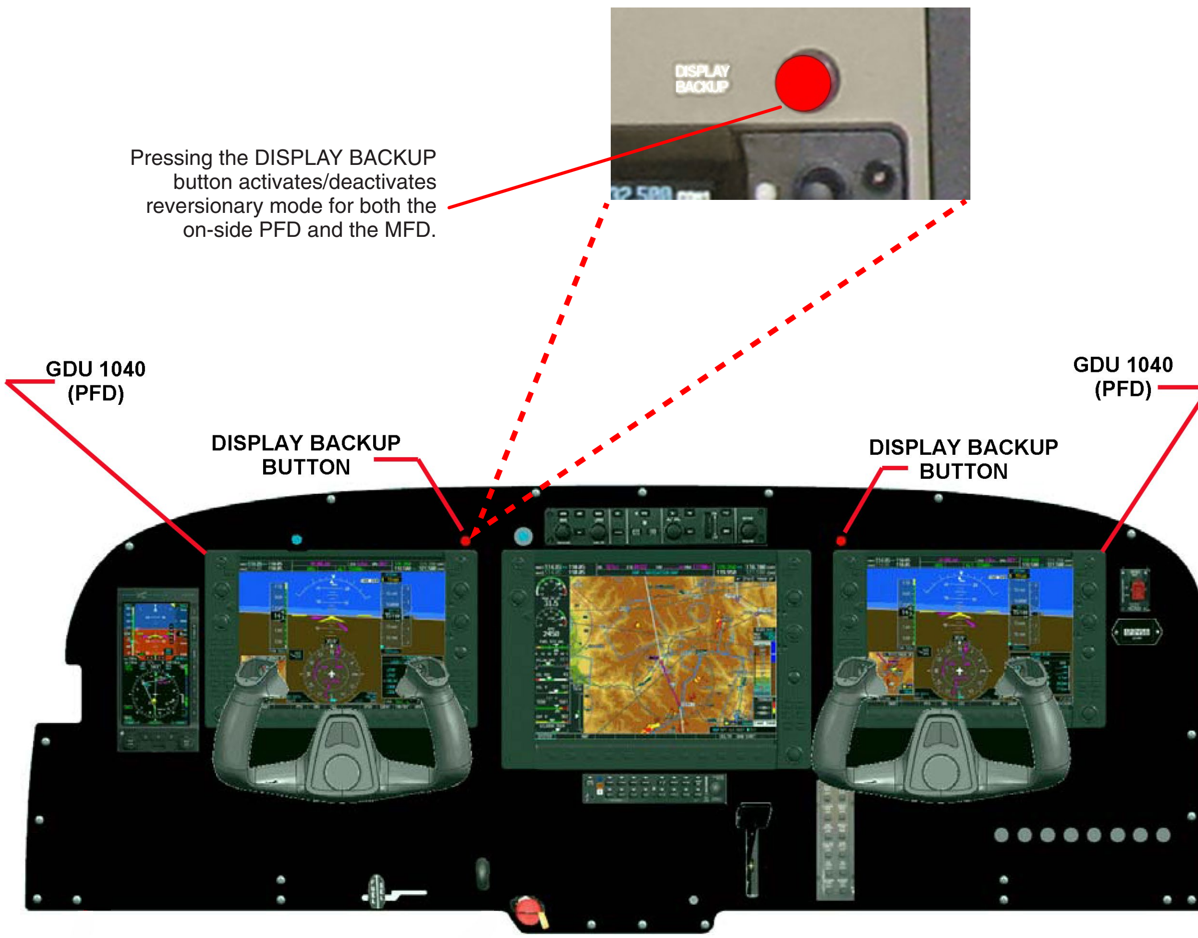
NOTE: The G1000 system alerts the pilot when backup paths are utilized by the LRUs. Refer to the Appendices in the Garmin Pilot's Guide for further information regarding system-specific alerts.

Reversionary mode is a mode of operation in which all important flight information is presented identically on at least one of the remaining displays (see "Figure 9" on page 34250114 and "Figure 10" on page 34250115). Transition to reversionary mode should be straight-forward for the pilot, flight parameters are presented in the same format as in normal mode. In the event of a PFD failure, the G1000 system automatically switches to reversionary (backup) mode. In reversionary mode, all important flight information is presented on the remaining display(s) in the same format as in normal operating mode.

- PFD1 failure – MFD enters reversionary mode; PFD2 remains in normal mode.
- MFD failure – No automatic switch to reversionary mode. Either PFD may be put into reversionary mode by manually pressing the DISPLAY BACKUP button above the respective PFD (see "Figure 11" on page 34250117).
- PFD2 failure – MFD enters reversionary mode; PFD1 remains in normal mode. Reversionary mode can be activated manually by pressing the dedicated DISPLAY BACKUP button above PFD1. Pressing this button again deactivates reversionary mode.

Each display can be configured to operate in reversionary mode, as follows:

- PFD1 – By pressing the DISPLAY BACKUP button above the left PFD.
- MFD – By pressing the DISPLAY BACKUP button above the left or right PFD.
- PFD2 – By pressing the DISPLAY BACKUP button above the right PFD.



Display Backup Button (Typical)
Figure 11

D. Troubleshooting

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 341014.

Troubleshoot the G1000 system by first identifying, then isolating the specific failure to the responsible LRU. There are several indications that the G1000 presents to the pilot or technician, showing overall system condition. A course of action should be determined based on the information presented on the display.

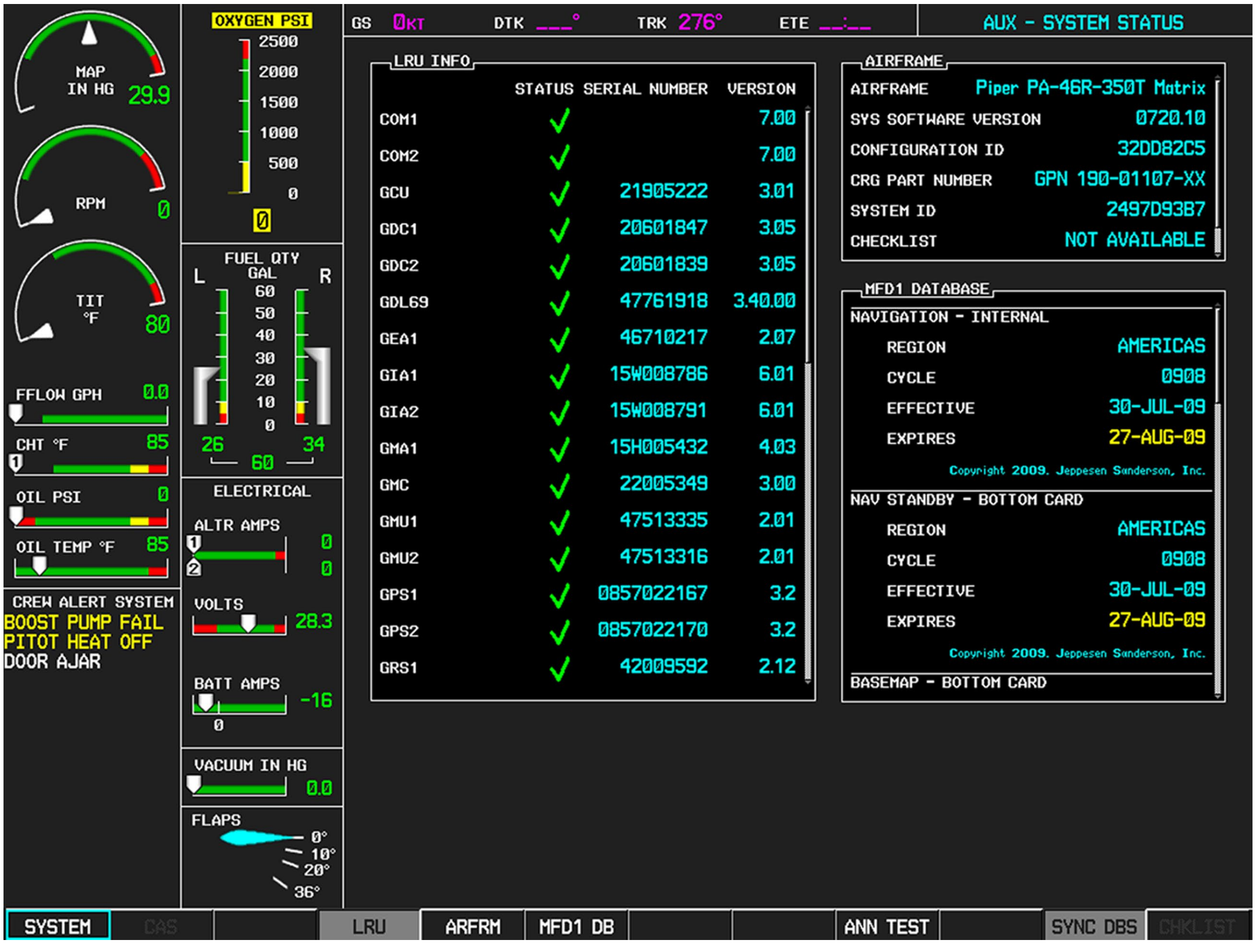
The 5th AUX group page on the MFD (see "Figure 12" on page 34250119) provides LRU health status by means of a green check or a red 'X'. Also, LRU software versions are shown along with other database versions and dates. If a red 'X' is shown for an LRU, see the appropriate LRU troubleshooting chart for guidance.

Typically, the G1000 Alerting System provides alerts/annunciations in conjunction with the information presented at the 5th AUX page.

CAUTION: "POST-INSTALLATION PROCEDURES" ON PAGE 342501171 PROVIDE DETAILED INSTRUCTIONS ON EQUIPMENT CONFIGURATION AND RETURN-TO-SERVICE TESTING. ANYTIME A G1000 COMPONENT OR LRU IS REMOVED, SWAPPED, OR REPLACED, THE TECHNICIAN MUST FOLLOW THE PROCEDURES GIVEN IN "POST-INSTALLATION PROCEDURES" ON PAGE 342501171 TO ENSURE PROPER OPERATION OF THE SYSTEM. SEE ALSO RETURN-TO-SERVICE UNDER THE INDIVIDUAL G1000 COMPONENT IN "COMPONENTS" ON PAGE 34250162.

Troubleshooting information is provided as follows:

(1) Line Replaceable Unit (LRU) Failures - see "Chart 5" on page 34250131.
(2) Engine / Airframe Sensor Failures - see "Chart 6" on page 34250137.
(3) TAWS - see "Chart 8" on page 34250143.
(4) GDU 1040 - see "Chart 9" on page 34250145, "Chart 13" on page 34250164, and "Chart 14" on page 34250166.
(5) GDU 1240A - see "Chart 9" on page 34250145, "Chart 13" on page 34250164, and "Chart 14" on page 34250166.
(6) GDU 1500 - see "Chart 9" on page 34250145, "Chart 13" on page 34250164, and "Chart 14" on page 34250166.
(7) GCU 476 - see "Chart 15" on page 34250176.
(8) GMA 347 - see "Chart 16" on page 34250180 and "Chart 17" on page 34250182.
(9) GMA 350 - see "Chart 16" on page 34250180 and "Chart 17" on page 34250182.
(10) GIA 63W - see "Chart 18" on page 34250186 and "Chart 19" on page 34250188.
(11) GEA 71 - see "Chart 20" on page 34250198.
(12) GTX 33 - see "Chart 21" on page 342501101.
(13) GDC 74A - see "Chart 22" on page 342501103.
(14) GTP 59 - see "Chart 5" on page 34250131.
(15) GRS 77 - see "Chart 23" on page 342501108, "Chart 24" on page 342501109, and "Troubleshooting" on page 342501105 under GRS 77 in Components.
(16) GMU 44 - see "Chart 23" on page 342501108 and "Chart 24" on page 342501109.
(17) GDL 69/69A - see "Chart 25" on page 342501115 and "Chart 26" on page 342501116.
(18) GSR 56 - see "Chart 27" on page 342501119.
(19) GWX 68 - see "Chart 28" on page 342501121.
(20) GTS 825 - see "Chart 29" on page 342501128.



System Status Page (AUX Group)
Figure 12

(21) GMC 710 - see "Chart 2" in 22-10-00.

(22) Software Configuration - see "Chart 31" on page 342501147.

(23) Synthetic Vision System - see "Chart 33" on page 342501167 and "Chart 34" on page 342501167.

(24) GFC 700 AFCS

See "Chart 7" on page 34250140.

The GFC 700 is a digital Automatic Flight Control System (AFCS) which is integrated into various components of the G1000. This section touches upon key items to note while troubleshooting the GFC 700.

Should a problem be encountered during the operation of the GFC 700, the pilot and technician should first evaluate the overall status and condition of the G1000 system at the AUX – System Status page (on MFD). Any alert messages, annunciations, or other abnormal behaviors should be noted in an effort to pinpoint the fault. The object is to locate the fault within a LRU or LRUs in efforts to replace the defective equipment.

The GFC 700 AFCS Annunciation field is located above the airspeed tape on the PFD as shown in "Chart 7" on page 34250140.

(25) GFC Status Page

The GFC Status page is presented in configuration mode and gives status information regarding the GFC 700.

See "Figure 13" on page 34250121.

(a) GIA Status

AP Disconnect: Shows the condition of the AP DISC +28 VDC input to the GIAs and servos, which is required for the Autopilot to operate. A green status indicator shows the AP DISC switch is closed and the GFC 700 is actively receiving 28. volts. A black indicator box indicates the GIAs and servos are no longer receiving the +28 VDC AP DISC power (switch open or other fault).

(b) Monitor/Control Board Status

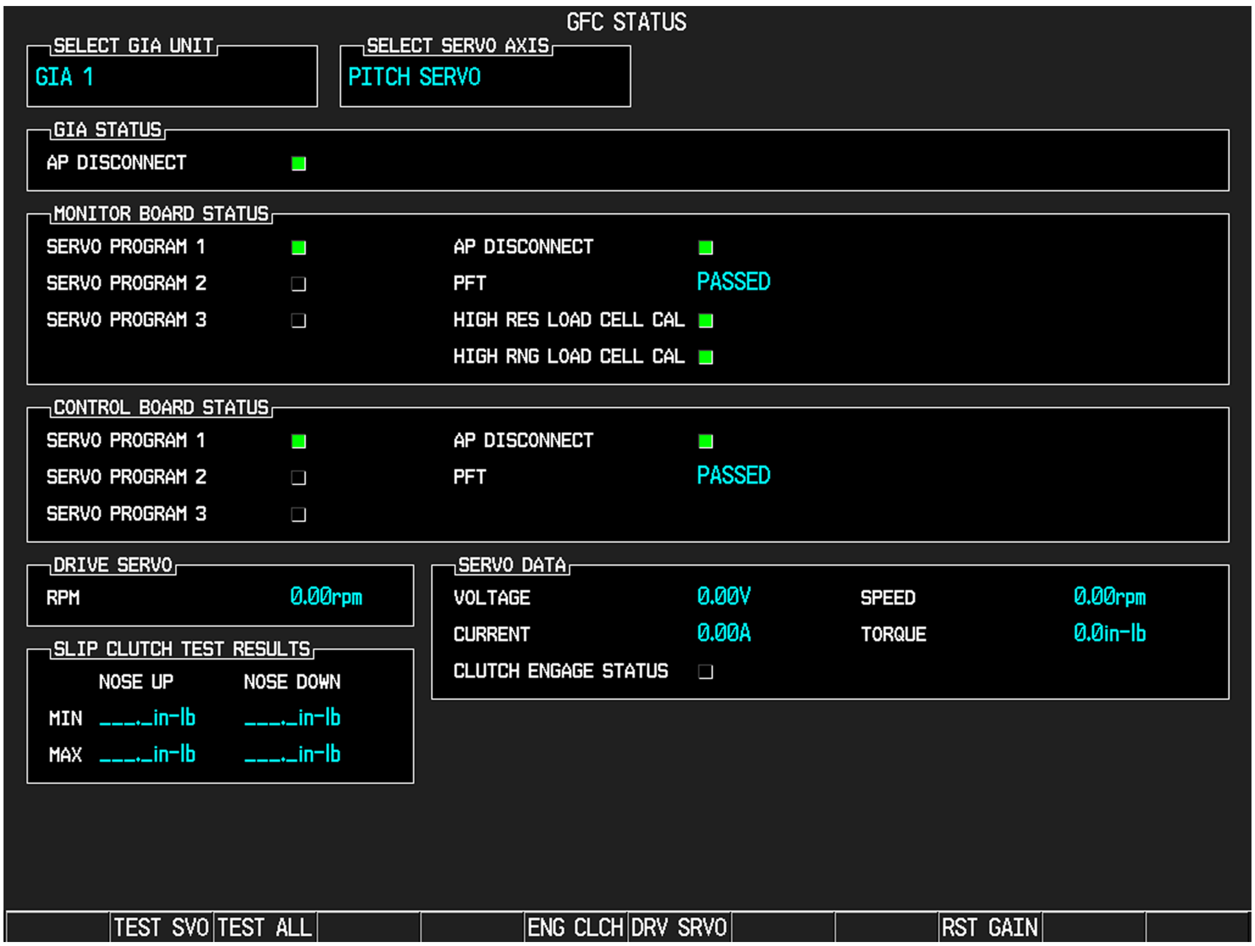
Shows the condition of various monitor board components.

(c) Servo Program (1-3): Servo program discretes are used to determine the HW strapping for each GSA to define the servo type. This information can be cross-referenced against the system interconnects to verify proper servo grounding.

(d) AP Disconnect: Same as GIA Status.

(e) PFT: Indicates whether the pre-flight test has passed or failed.

(f) HIGH RES & HIGH RNG LOAD CELL CAL: Shows the condition of the high resolution and high range load cells on the monitor board. If box is black, this indicates a corrupt or missing load cell calibration; return the servo to Garmin.



GFC Status Page
Figure 13

(26) Drive Servo

Allows the technician to enter a desired RPM at which to manually drive the selected servo. Direction of rotation is controlled by the polarity of the RPM (+ or -). After the speed is entered, the technician may use the ENG CLCH and DRV SRVO softkeys to drive the servo.

NOTE: Be especially certain that the flight controls are clear and safe to operate before manually driving the servo.

SERVO DATA

Shows real-time reported data including servo voltage, speed, motor current, load cell torque, and clutch solenoid status. A green box indicates the servo clutch is engaged.

(27) GIA Fault Log Descriptions

The section was created to help determine why the GFC 700 has failed the Pre-Flight Test (PFT) indicated by the red PFT annunciation, it defines the PFT sequence for the servos and the GFC 700 system and then provides troubleshooting information to help resolve failures.

There are 16 steps to the GFC 700 PFT. The PFT is performed by both GIA's at startup, and needs to pass on both GIA's before the autopilot can be engaged.

The PFT is only started if the AHRS has aligned, the GIA's and servos are configured and the certification gains are valid. If the PFT has not completed after one minute from when the initialization started, the PFT will fail. After the system PFT has passed, it will be performed again if a servo resets, if the autopilot servo breaker is reset or the cross side GIA restarts it.

Generally, the PFT failure fault is logged in the GIA Maintenance Log and not in the Servo Maintenance Logs unless the GIA log fault identifies a servo problem.

NOTE: Thoroughly understanding the operation of the G1000 system in Configuration mode is recommended before starting this procedure. The GFC Status page may be used to check the status of the servos and engage them to aid in troubleshooting.

(a) To access the GIA and GSA Maintenance Logs, perform the following steps:

1) Start the G1000 in Configuration mode.
2) Use the FMS knob on PFD1 to go to the Diagnostics Terminal page in the System group. This page allows the technician to view maintenance logs associated with the GFC 700.
3) Choose 'GIA1' or 'GIA2' in the LRU window.
4) In the SERVO window, choose 'NONE' to view the GIA Maintenance Log, or choose a servo to view their logs.
5) Using the FMS knob, choose 'VIEW MAINTENANCE LOG' in the COMMAND window.
6) Press the ENT key.
7) When the Maintenance Log data starts to display in the OUTPUT window, you may see "More...press any key to continue..." at the bottom of the OUTPUT window. This informs you there is more data to display and the system has paused allowing you to view the data before continuing. To see more of the data, reselect the "VIEW MAINTENANCE LOG" in the COMMAND window and press the ENT key. The "...press any key to continue..." function is not active at this time.
8) Scroll through the OUTPUT list by pressing the OUTPUT softkey.

(b) The GIA Maintenance Log can record any of the following faults:

1) FCS Task not started: Bad gains.

   The FCS task has not started because the gains are not present or have been corrupted.

   Reload the gain files to correct.

2) FCS Task not started: Gain structure out of range.

   The FCS task has not started because the gains are not compatible with the GIA software. Reload the gain files to correct.

3) PFT FAIL: Timeout, <STEP>

   Pre-flight test has failed because the specified step has not passed in the allotted time. See the GIA pre-flight steps for a description of the possible values for <STEP> on the failed GIA and corrective actions.

4) PFT FAIL: Cross GIA Failed, State: <STEP>

   Pre-flight test has failed on opposite GIA. <STEP> specifies the pre-flight test step on selected GIA that was in progress when the pre-flight test failed on the opposite GIA. See the GIA pre-flight steps for a description of the possible values for <STEP> on the failed GIA and corrective actions.

5) PFT FAIL: <STEP>

   Pre-flight has failed because the step specified has failed. See the GIA pre-flight test steps for a description of the possible values for <STEP> on the failed GIA and corrective actions.

6) AHRS MON invalid: <STATE>

   The AHRS monitor has detected that the AHRS data is invalid. The possible values for <STATE> are:

   a) Mon Prmtr Invalid: The ARINC 429 data used by one of the monitors has not been received.
   b) Attitude Prmtr Invalid: The ARINC 429 pitch or roll angle has not been received.
   c) Exceeded Attitude Limits: The pitch or roll angle has exceeded the engagement limits.
   d) Cross Hdg Accel Fail: Cross heading acceleration monitor failed.
   e) Vert Accel Fail: Vertical acceleration monitor failed.
   f) Fltrd Cross Hdg Accel Fail: Filtered cross heading acceleration monitor failed.
   g) Fltrd Vert Accel Fail: Filtered vertical acceleration monitor failed.
   h) Roll Accel Fail: Roll acceleration monitor has failed.
   i) Normal Accel Fail: Normal acceleration has failed.

7) Troubleshoot the GRS 77 for the cause of the failure.

8) Stuck switch invalidated parameter: <AXIS>

   A MET switch in the specified axis is stuck. Check the MET (trim) switches for proper operation.

9) PRMTR: <PARAMETER> MODE:<MODE> Parameter lost

The mode specified by <MODE> has been disengaged because the parameter specified by <PARAMETER> has become invalid. The following is a list of some of the possible values for <PARAMETER>:

10) AD TDM Comm Valid: The specified mode has been disengaged because communication with the servos, via the Time Division Multiplexer protocol, has been lost.

11) AP Pitch MET not stuck: The specified mode has been disengaged due to a stuck pitch MET switch.

12) Check the MET (trim) switches for proper identification.

(c) GIA Pre-Flight Test (PFT) Steps

0) PFT step 0: System initialization, verify GFC powered.

This step is checking to make sure the GFC is powered up. It checks to make sure the GIA AP disconnect input is connected to 28 volts, and makes sure the Servos are up and communicating. If this step fails, make sure the GIA is connected to AP disconnect by using the GFC configuration page. Also make sure all configured Servos are communicating by checking for Servo product data in configuration mode.

1) PFT step 1: System initialization, verify GIA audio is valid.

This step is checking to make sure that the GIA audio region has been loaded and configured. If this step fails, load GIA audio region and audio configuration.

2) PFT step 2: System initialization, verify required servos are configured.

This step is checking to make sure the current Servo configuration matches the Servo configuration specified in the certification gain file. If this step fails, then make sure the Servo configuration on the GFC configuration page matches the Servo configuration specified in the configuration gain (.cgn) file.

3) PFT step 3: System initialization, verify selected side.

This step is checking to make sure the PFD1 is online and sending the selected AFCS side data over.

HSDB to GIA1. If this step fails, then make sure the PFD is powered up and there is an Ethernet connection from the PFD to the GIA.

4) PFT step 4: System initialization, verify AHRS monitor

This step is checking to make sure the AHRS monitor is valid and not reporting an AHRS failure.

NOTE: AHRS monitor will be assumed valid if on the ground. If this step fails, then make sure the AHRS and ADC is powered up and sending valid attitude data to the G1000.

5) PFT step 5: System initialization, verify servo PFT is complete.

This step is checking to make sure that all servos have completed their own PFT. This does not check whether the servo PFT passed or failed. It verifies that the servo PFT is no longer in progress.

6) PFT step 6: Verify cross GIA is initialized.

This step is checking to make sure the other GIA is also on step 6 of its PFT. If this step fails, try cycling power on GIA1, GIA2, and all servos. If PFT still fails at step 6 then you will need to go check the PFT status of the other GIA and see what step it is failing (it should be prior to step 6).

NOTE: The PFT status is communicated between GIA1 and GIA2 using HSDB. As a result, both the PFD and MFD must be powered for this state to pass.

7) PFT step 7: Verify servo type.

This step is checking to make sure the Servos are the correct type (80 or 81, high or low torque). If this step fails, make sure the Servo configuration on the GFC configuration page matches the Servos installed in the aircraft.

8) PFT step 8: Verify servo first certification data.

This step is checking to make sure the servos have the same certification gains loaded in them as the GIAs have. If this step fails, reload the certification gains in GIA1, GIA2, and all servos.

9) PFT step 9: Verify servo second certification data.

This step is checking to make sure the servos have the same certification gains loaded in them as the GIAs have. If this step fails, reload the certification gains in GIA1, GIA2, and all servos.

10) PFT step 10: Updating servo RTC.

This step is setting the system time in the servos to by the same time as the GIA system time. This step should never fail.

11) PFT step 11: Verify servo PFT status.

This step is checking to make sure the servos have all passed their own PFT. If this step fails, please proceed to servo PFT explanation below.

12) PFT step 12: Verify AP disconnect enabled.

This step is checking to make sure GIA1, GIA2, and all servos have are connected to a 28 volt AP disconnect. If this step fails, make sure the AP disconnect input to GIA1, GIA2, and all servos is connected and registering 28 volts. Make sure the AP disconnect switch has been installed in the aircraft. Make sure no one is holding the AP disconnect switch down on the yoke.

13) PFT step 13: Verify servo validity.

This step is checking to make sure all the Servos are up and communicating with valid data to the GIA in TDM mode. If this step fails, then make sure all Servos are turned on and communicating by checking for green boxes on the GFC configuration page.

14) PFT step 14: Verify cross GIA PFT is completed.

This step is checking to make sure the other GIA is also on step 14 of its PFT. If this step fails, try cycling power on GIA1, GIA2, and all servos. If PFT still fails at step 14 then you will need to go check the PFT status of the other GIA and see what step it is failing.

NOTE: The PFT status is communicated between GIA1 and GIA2 using HSDB. As a result, both the PFD and MFD must be powered for this state to pass.

15) PFT step 15: PFT completed.

The PFT has successfully completed.

16) PFT step 16: PFT failed.

The PFT has failed.

(d) Servo Faults and Troubleshooting

Whenever a servo fault occurs, a status message is logged to the corresponding servo control or monitor maintenance log. This information is also accompanied by a time and date stamp. An "RTC DATE" entry is made every time a servo is powered on, it is normally not useful for troubleshooting.

The following is a listing of possible faults that could be reported in a GSA fault log. Faults can occur in either the monitor board processor or the control board processor, both of which are contained in the GSA unit.

1) Monitor Processor

The monitor processor contains the logs that are found in these processors:

2 – Pitch Servo

4 – Roll Servo

6 – Yaw Servo

8 – Pitch Trim Servo

There are two main groupings of faults that can occur in the monitor processor:

a) The first grouping of faults can occur during the GSA unit pre-flight test (PFT). If there is a fault during PFT the unit will not be able to transition to normal mode and the only way to clear this state would be to cycle unit power. See "Chart 1" on page 34250127.

b) The second grouping of faults can occur during normal mode. These faults generally cause a disconnect of power to the GSA and report that a fault has occurred to the GIA. The Notes column indicates any actions that can be taken to troubleshoot the problem in the aircraft by the technician. Any faults that are not listed here indicate an internal problem requiring replacement of the servo. If the items in the Notes column check out ok, replace the servo. See "Chart 2" on page 34250127.

2) Control Processor

The control processor contains the logs that are found in these processors:

3 – Pitch Servo

5 – Roll Servo

7 – Yaw Servo

9 – Pitch Trim Servo

There are two main groupings of faults that can occur in the control processor.

a) The first grouping of faults can occur during the GSA unit pre-flight test (PFT). If there is a fault during PFT the unit will not be able to transition to normal mode and the only way to clear this state would be to cycle unit power. See "Chart 3" on page 34250128.

b) The second grouping of faults can occur during normal mode. These faults generally cause a disconnect of power to the GSA and report that a fault has occurred to the GIA. See "Chart 4" on page 34250128.

The Notes column indicates any actions that can be taken to troubleshoot the problem in the aircraft by the technician. Any faults that are not listed here indicate an internal problem requiring replacement of the servo. If the items in the Notes column check out OK, replace the servo.

CHART 1
MONITOR PROCESSOR - PFT FAULTS

MONITOR PFT STEP	NOTES
"INTERNAL COMM FAIL"	This can sometimes be a result of a failure on the other internal servo board, check faults on the other processor
"UNSW POWER INV"	Check unit power
"MON SOL PWR ON FAIL"	Check unit power and AP Disconnect power
"CTL SOL PWR ON FAIL"	Check unit power and AP Disconnect power
"SOL PWR FAIL"	Check unit power and AP Disconnect power
"CERT DATA UNINSTALLED"	Upload the certification gain file to the Monitor board
"STRAP CODE MISMATCH"	Check the connector strap inputs to the unit

CHART 2
MONITOR PROCESSOR - NORMAL MODE FAULTS

MONITOR PFT STEP	NOTES
"INTERNAL COMM FAIL"	This can sometimes be a result of a failure on the other internal servo board, check faults on the other processor
"UNSW POWER INV"	Check unit power
"MON SOL PWR ON FAIL"	Check unit power and AP Disconnect power
"CTL SOL PWR ON FAIL"	Check unit power and AP Disconnect power
"SOL PWR FAIL"	Check unit power and AP Disconnect power
"CERT DATA UNINSTALLED"	Upload the certification gain file to the Monitor board
"STRAP CODE MISMATCH"	Check the connector strap inputs to the unit

(28) Downloading GIA and GSA Maintenance Logs.

If additional assistance is needed troubleshooting autopilot faults, the Maintenance logs can be downloaded to an SD card as a text file (.txt) and emailed to Garmin Aviation Product Support. Please call Garmin Aviation Product Support before you send a Maintenance Log to notify them you are sending it to prevent a delay in response. You may download multiple GIA and GSA Maintenance Logs to the same file, however in your email to Garmin you must furnish the order in which they were downloaded (i.e. GIA1, then GIA2, then SRVO PTCH MON, then SRVO PTCH CTL, etc.).

(a) Insert a FAT 32 formatted SD card into the top slot of the PFD before turning on the displays.

(b) Power up PFD1/2 and MFD in the configuration mode.

(c) On the PFD1 in the System page group, use the small FMS knob to scroll to the Diagnostics Terminal page.

(d) Press the LG2CRD softkey at the bottom of the PFD1. Verify that the softkey text grays out. This indicates the recording function is active and all text that is displayed in the OUTPUT window will be saved to the card.

CHART 3
CONTROL PROCESSOR - PFT FAULTS

CONTROL PFT STEP	NOTES
"INT COMM TEST FAIL"	This can sometimes be a result of a failure on the other board, check faults on other processor
"CTL MOT PWR ON FAIL"	Check unit power and AP Disconnect power
"MON MOT PWR ON FAIL"	Check unit power and AP Disconnect power
"HALL 1 FAIL"	Check unit power and AP Disconnect power
"HALL 2 FAIL"	Check unit power and AP Disconnect power
"HALL 3 FAIL"	Check unit power and AP Disconnect power
"HALL 4 FAIL"	Check unit power and AP Disconnect power
"HALL 5 FAIL"	Check unit power and AP Disconnect power
"HALL 6 FAIL"	Check unit power and AP Disconnect power
"CURR OFFST FAIL"	Check unit power and AP Disconnect power
"SVO TYPE FAIL"	Check unit power and AP Disconnect power
"CERT DATA UNINSTALLED"	Upload the certification gain file to the Control board
"STRAP CODE MISMATCH"	Check the connector strap inputs to the unit

CHART 4
CONTROL PROCESSOR - NORMAL MODE FAULTS

CONTROL FAULT	NOTES
"GIA DIS FAULT"	Check the AP Disconnect power into the unit
"HOST DATA DIF"	Check the AHRS wiring to the system
"HOST DATA INV"	Check the AHRS wiring to the system
"SVO PWR INV"	Check unit power and AP Disconnect power
“STRP CODE CHNG”	Check the connector strap inputs to the unit
“MET STUCK SWTCH”	Check the MET switch inputs into the system
“MET STATUS DIF”	Check the MET switch inputs into the system

(e) Enable the curser by pressing the FMS knob, select “GIA1” in the LRU drop down menu and then press the ENT key to select it.

(f) Skip the SERVO box and move the cursor to the COMMAND box and select “View Maintenance Log” in the drop down menu then press the ENT key. The error log data will be displayed in the OUTPUT box. If you see the “more…press any key to continue” text at the bottom of the screen, you may need to reselect “View Maintenance Log” for GIA data to allow it to continue scrolling down the screen (pressing any key will not continue, disregard the text instruction you to do so). Continue to scroll through all the OUTPUT data until you see the text, “End of Fault Log”.

(g) Move the curser back to the LRU box, select “GIA2” in the LRU drop down menu and then press the ENT key to select it.

(h) Skip the SERVO box and move the cursor to the COMMAND box and select “View Maintenance Log” in the drop down menu then press the ENT key. The error log data will be displayed in the OUTPUT box. If you see the “more…press any key to continue” text at the bottom of the screen, you may need to reselect “View Maintenance Log” for GIA data to allow it to continue scrolling down the screen (pressing any key will not continue, disregard the text instruction you to do so). Continue to scroll through all the OUTPUT data until you see the text, “End of Fault Log”.

(i) If you need to download Servo fault logs (usually done at the request of Garmin Product Support), perform the following steps. Otherwise, skip to step (j).

1) In the LRU box, you may select either “GIA1” or “GIA2”.

2) In the SERVO box, choose a servo using the FMS knobs. Each servo contains two logs, one in the Monitor (MON) processor and one in the Control (CTL) processor. You must download both for each servo separately.

3) In the COMMAND box, select “View Maintenance Log” and press the ENT key.

4) The log will appear in the OUTPUT box. It will scroll to the end automatically. When it is complete, repeat steps 1) - 3) for the other servos in the aircraft. Be sure to note the order the servos were downloaded in including the Monitor or Control logs to email to Garmin Product Support. Without knowing the order in which the logs were downloaded, Garmin will be unable to process them and will ask for another full download.

(j) Press the LG2CRD softkey to turn off the recording function.

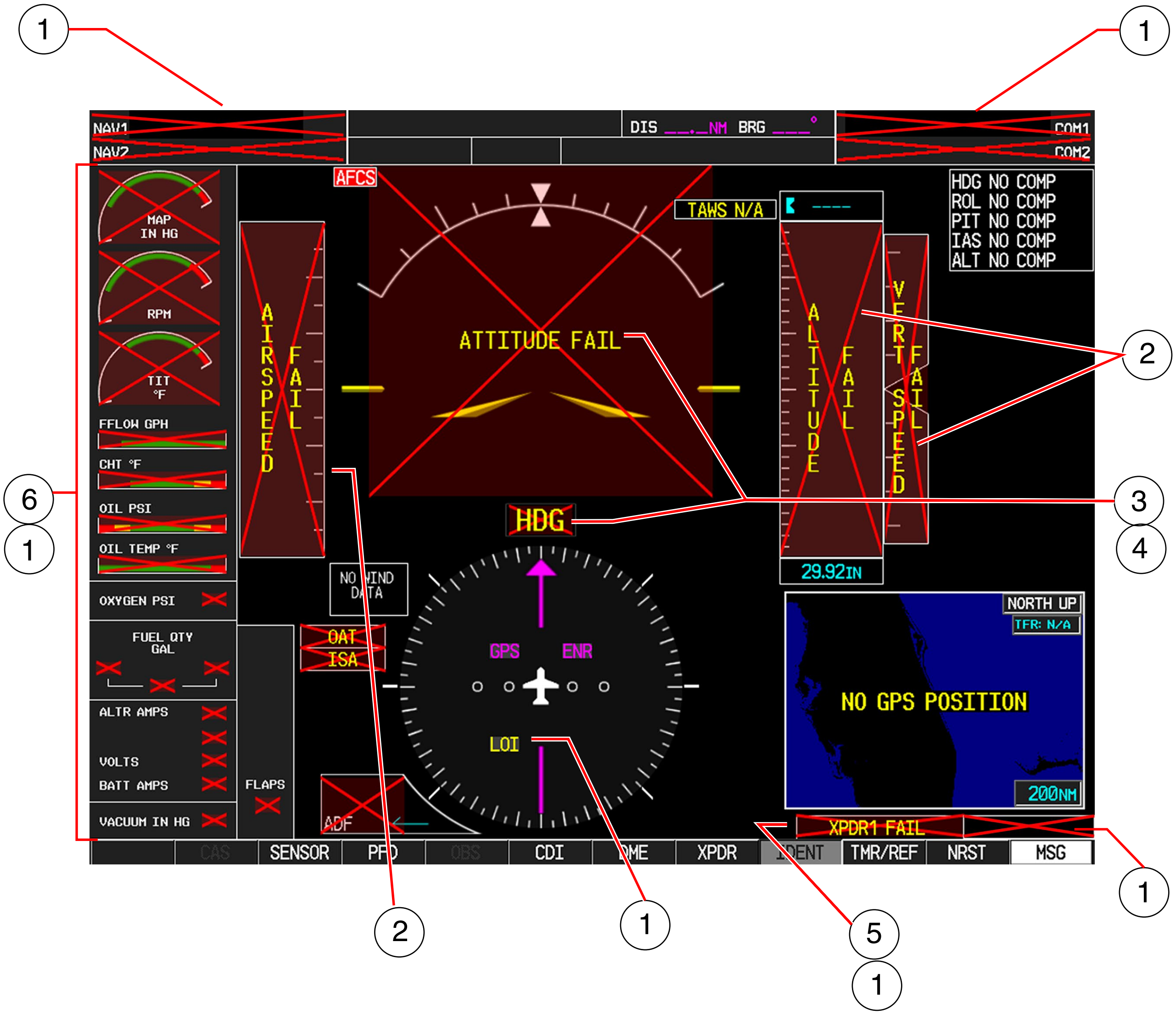
(k) Wait 1 minute for the system to save the data from the download to the SD card.

(l) While you are waiting for the data to be saved to the SD card, record the order of the LRU’s and/or Servos were downloaded so that you can provide that information to Garmin to help decipher the order of the error data.

(m) Power down the G1000 System and remove the SD card.

(n) Insert the SD card in the card reader of a laptop or desktop computer and open the “diag_buf_log.txt” file from the SD card using the WordPad program. Verify that all of the fault logs were downloaded by checking for the “End of Fault Log” message at the end of the GIA data, and that the last servo log entry has the current date.

(o) Insert the fault log as an attachment to an email and include the LRU order how the data was downloaded and send to Garmin Aviation Product Support at avionics@Garmin.com.



1. GIA 63W Integrated Avionics Units
2. GDC 74A Air Data Computer
3. GRS 77 AHRS
4. GMU 44 Magnetometer
5. GTX 33 Transponder
6. GEA 71 Engine Airframe Units

System Fail
Figure 14

CHART 5 (Sheet 1 of 6)
LRU TROUBLESHOOTING

System Failure Troubleshooting

The following table provides basic troubleshooting guidance for LRU failures.

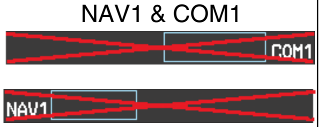
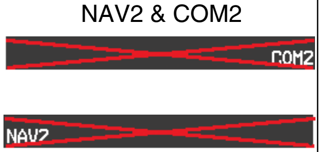
Annunciation	Associated LRU(s)	Solution
Low Speed Awareness Band permanently displayed 	GDU	• With aircraft weight on wheels and the G1000 in Configuration Mode, check that the **GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is illuminated. • If indication is illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace defective unit. ✓ If indication is not illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
Low Speed Awareness Band permanently inhibited 	GDU	• With aircraft weight off wheels and the G1000 in Configuration Mode, check that the **GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is not illuminated. • If indication is not illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace unit. ✓ If indication is illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
NAV1 & COM1 	GIA1	• Check configuration settings for GIA1 and PFD1. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA1, replace GIA1. • Check Ethernet interconnect from GIA1 to PFD1. ✓ If problem persists, replace PFD.
NAV2 & COM2 	GIA2	• Check configuration settings for GIA2 and PFD2. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA2, replace GIA2. • Check Ethernet interconnect from GIA2 to PFD2. ✓ If problem persists, replace PFD.

CHART 5 (Sheet 2 of 6)
LRU TROUBLESHOOTING

Annunciation	Associated LRU	Solution
GPS LOI, INTEG or DR  	GIA1 or 2	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check PFD1 Alert Window for GIA1/2 configuration, software or failed data path error messages. Correct any errors before proceeding. • Verify the aircraft is located where the GPS antennas have a clear view of the sky. • Verify the aircraft is not parked in close proximity to a hanger with the doors open equipped with a GPS repeater. • On the MFD AUX-GPS STATUS page, check for erratic GPS Signal Strength bars. If they are erratic, external interference is affecting the GPS receiver. Locate source of interference and remove. • Swap GIA1 and GIA2 to verify location of problem. If problem follows the GIA, replace the GIA. • Check corresponding GPS antenna and cable for faults. Correct antenna or cable fault. • Check PFD1 to GIA1 and PFD2 (or MFD) to GIA2 Ethernet interconnect for faults. Correct interconnect fault. • If problem persists; replace PFD1, PFD2 or the MFD that shows the problem.

CHART 5 (Sheet 3 of 6)
LRU TROUBLESHOOTING

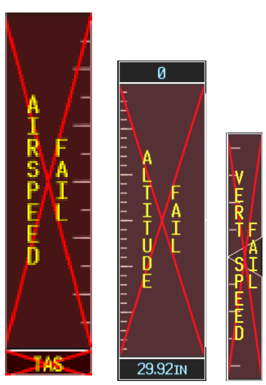
Annunciation	Associated LRU	Solution
XPDR FAIL 	GTX 33	• Check PFD Alert Window for GIA1/2 and GTX 33 configuration, software or failed data path error messages. Correct any errors before proceeding. • Perform a SET>ACTV configuration reset on the GTX Transponder Configuration page for each installed GTX. • For GTX 33 transponders verify the aircraft registration is entered in the GTX Transponder Configuration page. • Check the GIA and GTX racks for connector pin faults (push-back or bent) on the RS-232 interconnect lines. • Replace the GTX 33.
TAS FAIL AIRSPEED FAIL, ALTITUDE FAIL, VERT SPEED FAIL 	GDC 74A	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check PFD1 Alert Window for PFD1/2, MFD or GDC configuration, software or failed data path error messages. Correct any errors before proceeding. • Inspect GDC 74A pitot/static ports and plumbing for blockage. • Check GDC 74A configuration settings for the PFDs, MFD, GIA1, and GIA2. Reload if unsure they are correct. • If PFDs, MFD, and GIA configuration settings are correct, replace the GDC 74A. • If problem persists, replace the GDC 74A configuration module.
OAT 	GTP 59	• Check OAT probe wiring, probe and connectors for faults or damage. • Replace GDC 74A config module and pigtail harness. Replace the GTP 59. • If problem remains, replace GDC 74A.

CHART 5 (Sheet 4 of 6)
LRU TROUBLESHOOTING

Annunciation	Associated LRU	Solution
HDG FAIL 	GRS 77 & GMU 44	• Check PFD1 Alert Window for PFD1/2, MFD or GRS configuration, software or failed data path error messages. Correct any errors before proceeding. • Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements. • Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power after moving aircraft away from metal objects to determine if metal objects were the source of the interference. Allow up to five minutes for the heading to reinitialize. • Perform a Magnetometer Interference Test to check for interference from onboard electrical system components (e.g. NAV lights). Pay particular attention to any new electrical devices that have been installed since the aircraft was new. Correct any discrepancies that do not allow this test to pass before continuing. • Ensure GRS 77 and GMU 44 connectors are secure. • Check the wiring and any inline connectors between the GRS and GMU for faults. • Recalibrate the GMU 44. • Replace the GMU 44. • If problem persists, replace the GRS 77.
	GRS 77 & GMU 44	• If this message persists longer than five minutes, perform AHRS calibration procedures as described later in this chapter.

CHART 5 (Sheet 5 of 6)
LRU TROUBLESHOOTING

Annunciation	Associated LRU	Solution
ATTITUDE FAIL 	GRS 77	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements. • Check PFD1 Alert Window for PFD1/2, MFD or GRS configuration, software or failed data path error messages. Correct any errors before proceeding. • Ensure GRS 77 unit connecter is secure and proper wire harness strain relief is provided. • Ensure the GRS 77 is fastened down tightly in its mounting rack and that the mounting rack is not loose (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Cycle GRS 77 power to restart initialization. • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important for an ATTITUDE FAIL that appears during ground operation only. • Perform an Engine Run-Up Test to check if engine vibration is causing the GRS 77 to go offline. • Replace GRS 77. • If problem persists, replace the GRS77 configuration module. • Contact Garmin Aviation Product Support if condition continues after replacing the GRS 77 and config module for additional assistance.

CHART 5 (Sheet 6 of 6)
LRU TROUBLESHOOTING

A DME or ADF failure is represented by the following red X's. The PFD displays function as a control head for the remote-mounted ADF and DME.

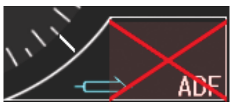
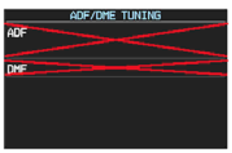
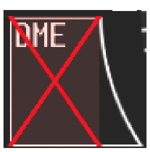
Annunciation	Sensor	Solutions
 	ADF	• Ensure that GIA 63 #2 is properly functioning. • Reload the ADF option configurations. • Check for proper operation of the ADF receiver. Ensure that the receiver is receiving power. • Check ADF – GIA2 interconnect.
 	DME	• Ensure that GIA 63 #2 is properly functioning. • Reload the DME option configurations. • Check for proper operation of the DME receiver. Ensure that the receiver is receiving power. • Check DME – GIA2 interconnect.

CHART 6 (Sheet 1 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

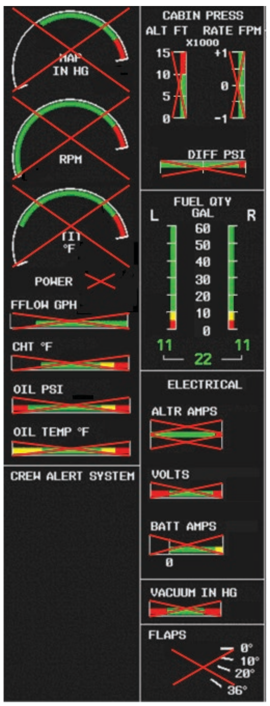
Invalid Data Field	Associated LRU(s)	Solution
Engine/Airframe Sensors (All Invalid)  (Typical)	GEA 71 & GIA 63W	• If software was loaded to a new GDU display, be sure that the user settings for the replaced display were cleared. Clear user settings by pressing the CLR key on the replaced display while applying power to it. Acknowledge the on-screen prompt by pressing the ENT key or the right-most softkey. • Check for GEA related Alert messages on the PFD. Correct any Alerts concerning software or configuration errors by reloading software or configuration as noted. • Verify GEA internal power supply, configuration, and calibration status in configuration mode. ✓ The internal power supply, configuration and calibration boxes should be green. If they are red, replace the GEA 71. • Verify internal, external, and reference voltages listed in the Main Analog and I/O A Analog boxes are not dashed out (does not include Aircraft Power 1 and 2). ✓ If any voltages are dashed out, replace the GEA. • Ensure the GEA is online (green checkmark on the AUX – SYSTEM STATUS page). ✓ If GEA is not online verify unit is receiving power at the rack connector. ✓ Check the GIA/GEA interconnects for faults. ✓ Reload configuration files to both GIA's and the GEA. • If problem persists, replace the GEA 71.

CHART 6 (Sheet 2 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Engine/Airframe Instrument Failures

The following table provides guidance for troubleshooting individual engine/airframe sensor failures. Be sure to also follow previous guidance given for the GEA 71. The technician should troubleshoot to isolate the fault by checking sensor-to-GEA wiring, replacing the suspect sensor, and finally by replacing the GEA 71. Replace one part at a time. Check for correct operation of the sensors and GEA 71 after any part has been replaced. Refer to G1000/GFC 700 in 91-00-00 Wiring Diagrams.

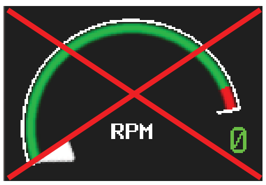
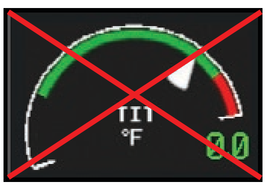
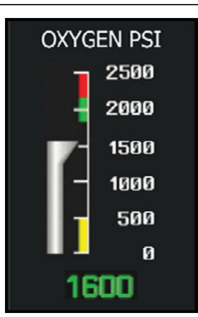
Invalid Field	Sensor	**Possible Solutions** (for applicable engine/system)
	Manifold Pressure	• Check manifold pressure transducer wiring. • Replace manifold pressure transducer wiring. • Replace GEA 71.
	Propeller RPM	• Check engine tachometer sensor wiring. • Replace engine tachometer sensor. • Replace GEA 71.
	Turbine Inlet Temperature	• Check TIT probe wiring. • Replace TIT probe sensor. • Replace GEA 71.
	Fuel Flow	• Check fuel flow transmitter wiring. • Check fuel flow transmitter - GEA wiring. • Check power input to fuel flow transmitter. • Replace fuel flow transmitter. • Replace GEA 71.
	Oil Pressure	• Check oil pressure sensor – GEA wiring. • Replace oil pressure sensor. • Replace GEA 71.
	Oil Temperature	• Check oil temperature sensor – GEA wiring. • Replace oil temperature sensor. • Replace GEA 71.
	Oxygen (Matrix Only)	• Check oxygen pressure transducer wiring. • Replace oxygen pressure transducer sensor. • Replace GEA 71.

CHART 6 (Sheet 3 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

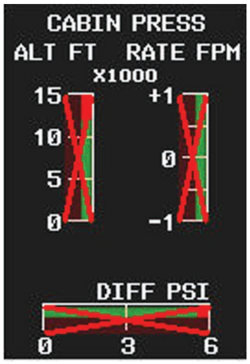
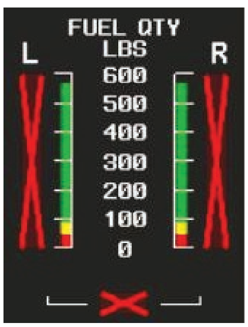
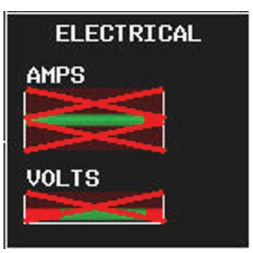
Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Cabin Pressure (Mirage Only)	• Check GAE 43 sensor – GEA 71 wiring. • Replace GAE 43 sensor. • Replace GEA 71.
	Differential PSI (Mirage Only)	• Follow cabin pressure steps. • Follow GDC 74A steps. • Check GEA 71 - GIA 63W wiring. • Check GDC 74A - GIA 63W wiring. • Replace GEA 71. • Replace GDC 74A.
	Fuel Quantity	• Check Fuel Senders - #1 GIA 63W wiring. • Check #1 GIA 63W - #1PFD wiring. • Check MFD - #1 PFD connections. • Replace Fuel Senders left or right • Replace #1 GIA 63W • Replace #1 PFD • For more details see 28-00-00.
	Amp/Volts	• Check Alt amp sensor - GEA 71 wiring. • Replace Alt amp sensor. • Replace GEA 71. • Check main bus - GEA 71 wiring. • Check emergency bus - GEA 71 wiring. • Replace GEA 71
	Vacuum (Mirage - Std) (Matrix - w/ Deice Opt)	• Check vacuum sensor - GEA 71 wiring. • Replace vacuum sensor. • Replace GEA 71.
	Flaps	• Check Potentiometer - GEA 71 wiring. • Replace Potentiometer. • Replace GEA 71.

CHART 7 (Sheet 1 of 3)
AFCS ANNUNCIATION TROUBLESHOOTING

GFC 700 ANNUNCIATION WINDOW

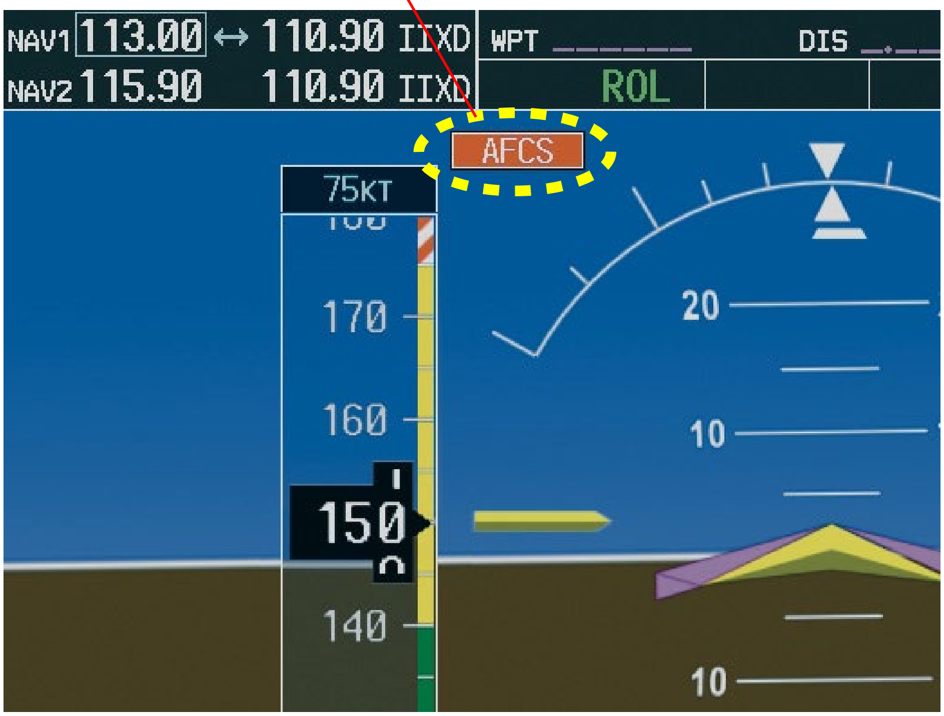


CHART 7 (Sheet 2 of 3)
AFCS ANNUNCIATION TROUBLESHOOTING

Condition	Annunciation	Recommended Actions
Pitch Failure	PTCH	• Check the AUX– SYSTEM STATUS page to see if the servo is online (green check). • Check that the affected servo is receiving power. • Check the servo wiring and connector. • Ensure PTRM switches are not stuck. • If failure condition still exists, remove and replace the affected servo.
Roll Failure	ROLL	
Yaw Damper Failure	YAW	
MEPT Sw. Stuck, or Pitch Trim Axis Control Failure	PTRM	
AFCS System Failure	AFCS	• Check that no red X's are present on the MFD and PFDs. • Check that there are no Alert Messages present in the PFD Alert window. Correct any software or configuration errors noted. • Go to the AUX SYSTEM STATUS page on the MFD and verify that all LRUs have a 'green' check. • Download GIA fault logs (see "GIA Fault Log Descriptions") and review for failure information. • If the OAT and TAS is Red-X'd and the attitude indication is present, troubleshoot per Chart 3.
Emergency Descent Mode (PA-46-350P)	EDM	• Decouple autopilot.
Elevator Mistrim Up	↑ELE	• If mistrim annunciations persist, check the Pitch and Pitch Trim servos for proper operation. Verify that the servo is online at the AUX - SYSTEM STATUS page. • Check the Pitch Trim servo wiring and connector. Ensure the servo is receiving power. • Check the aircraft trim control rigging. If mistrim condition still exists, remove and replace the affected servo.
Elevator Mistrim Down	↓ELE	
Aileron Mistrim Left	←AIL	• Check for possible fuel imbalance. • Check aileron control rigging. If mistrim condition still exists remove and replace the roll servo.
Aileron Mistrim Right	AIL→	
Rudder Mistrim Left	←RUD	• Check the AUX– SYSTEM STATUS page to see if the servo is online (green check). • Check that the affected servo is receiving power. • Check the servo wiring and connector. • If failure condition still exists, remove and replace the affected servo.
Rudder Mistrim Right	RUD→	
Preflight Test	PFT PFT	Reset system power. Allow the system to complete pre-flight tests. The preflight test should finish within 2 minutes. If it does not pass, the red 'PFT' annunciation is shown. In case of PFT failure, troubleshoot in the same manner as for the red 'AFCS' annunciation.

CHART 7 (Sheet 3 of 3)
AFCS GENERAL TROUBLESHOOTING

Condition	Resolution
Poor AP Performance (Limited AP Authority)	• Check each servo mount slip clutch per 22-10-00. Verify the clutches are not excessively loose and are within torque limits. • Check aircraft controls for proper balancing and adjustment per 55-20-00, 55-40-00, and 27-00-00. • Check aircraft control cables for proper tension per 22-10-00.
AP DISC Problems	• For intermittent nuisance disconnects with no AFCS or PFT alert, check A/P disconnect switch and wiring for intermittent faults. • If an AFCS or PFT alert is displayed at the time of the disconnect, troubleshoot per 22-10-00, Figure 9 and Post-Installation Procedures. • Check the GIA AFCS Fault Logs. If a **"Mon Prmtr Invalid"** message is received, check for valid true airspeed. A faulty GTP 59 OAT probe may cause TAS to become invalid, which will flag the "**Mon Prmtr Invalid**" message. • Contact Garmin Product Support for assistance.
Loss of Manual Electric Trim	• Check pitch trim servo status. • Check MET switch discrete inputs to both GIAs by going to the GIA I/O Configuration page and selecting DISCRETE IN inputs.
AutoTrim Inoperative	• If DATA indicator fails to illuminate or illuminates incorrectly, troubleshoot flap motor & discrete input wiring.

CHART 8
TAWS TROUBLESHOOTING

Annunciation	Cause	Solution
TAWS FAIL	A TAWS system failure has occurred.	• If message occurred on the first power up after unlocking TAWS, cycle power to initialize TAWS. • Ensure each GDU contains the correct database data card. • Ensure the G1000 is configured for TAWS: ✓ If the system is not configured for TAWS, reconfigure per 34-20-00, TAWS Unlock Procedure. • Verify GIAs are online. • Ensure a database or GDU SW mismatch has not occurred. ✓ If a mismatch has occurred, load correct database/software files or replace the terrain card.
TAWS TEST	TAWS system is currently being tested.	N/A – Test will be conducted up to two minutes.
TAWS INHB	TAWS system alerting is disabled	Enable TAWS system alerting by pressing the INHIBIT softkey on the MAP – TAWS page.
TAWS N/A	GPS system integrity not high enough to enable TAWS	• Ensure valid GPS position is received from the AUX – GPS STATUS page. • Check GPS antenna & associated coaxial cabling. • Troubleshoot GIAs.

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(29) Backup Communications Path Checks.

The G1000 system architecture is designed with redundant communication ports for several LRUs so that critical information can continue to be displayed in the event of an equipment or wiring failure. Of most importance is flight attitude, heading, and air data information. The GRS 77 and GDC 74A each have four separate ARINC 429 data lines which are all capable of sending data to the displays. The GEA 71, GTX 33/33D, and GMA 347 / GMA 350 each have two redundant serial communication paths for the same purpose. See "Figure 2" on page 3425014 for a basic G1000 block diagram depicting this architecture. When troubleshooting, refer to the G1000/GFC 700 Wiring Diagrams listed in Chapter 91-00-00.

(30) Failed Path Messages.

The following message indicates there is a data path connected to the GDU (PFD1, PFD2, or MFD) or the GIA 63W (1 or 2) that has failed.

FAILED PATH – A data path has failed.

The FAILED PATH message is triggered by a timeout of any one digital channel. The channels that are checked are listed on these pages in config mode:

(a) GDU RS-232 / ARINC 429 CONFIG (PFD1/2, and MFD).

(b) GIA RS-232 / ARINC 429 CONFIG (GIA1 and GIA2).

(c) GIA CAN / RS-485 CONFIGURATION (GIA1 and GIA2).

NOTE: Once the FAILED PATH message has been triggered, it will remain on the list of messages until the next power cycle. This latching was implemented so that for intermittent failures, the message would remain at the end of the flight (to alert maintenance crew). Also, this keeps the crew from having to acknowledge message repeatedly in the case of intermittent failures.

The box next to each channel indicates the current status of the channel per the below:

Red = data path is known to be failed.

Black = data path status is unknown.

Green = data path is known to be good.

The applicable data paths can be verified by viewing the configuration mode pages listed in "Chart 9", below, through "Chart 11" on page 34250147. See "Chart 30" on page 342501143 to navigate to a particular configuration page.

CHART 9
G1000 - TROUBLESHOOTING - DATA PATH FAILURES

Failure Message	Cause	Solutions
FAILED PATH – A data path has failed.	A communications data path status is not being received by the G1000.	Determine which data path has failed: See G1000 backup path test below. Check wiring continuity for the failed path. Swap or replace the affected LRU.

CHART 10
FAILED PATH MESSAGES, GDU RS-232 / ARINC 429 CONFIG PAGE (PFD1/2, AND MFD)

SELECT UNITS = MFD / PFD1 / PFD2 / GIA1 / GIA2 – ARINC-429 / RS-232 / RS-485 Serial Config Pages

Indicator	Status
	SELECT UNIT/LRU data path is functioning correctly.
	SELECT UNIT/LRU data path is not functioning correctly. • Check GTC avionics status page for SELECT UNIT or LRU configuration or software error messages. Correct any errors before proceeding. • If applicable, check the SELECT UNIT/LRU interconnect wiring and unit connector pins for faults. • Replace LRU. • Reload SELECT UNIT config file. • If problem remains, replace SELECT UNIT. • Remove SELECT UNIT and verify power and ground are present at the appropriate connector. ✈ If power or ground is not present, troubleshoot aircraft wiring for faults. ✈ If power and ground are present, check SELECT UNIT unit and LRU connector for damaged or pushback pins. • Swap PFD1 and PFD2 to confirm if the problem is in the original PFD1. • Replace original PFD1 if box displays green check after swapping displays. • Ensure LRU connecter is secure and proper wire harness strain relief is provided. • Swap LRU #1 and LRU #2 to confirm if the problem is in the original LRU #1. • Replace original LRU #1 if box displays green check after swapping units.
	SELECT UNIT/LRU data path functionally is unknown. • Reload SELECT UNIT configuration file.

CHART 11
DATABUS ACTIVITY CHECKS

UNIT	BUS ACTIVITY	INTERFACE
PFD1	RS 232 CH 1 IN/OUT GCU476	1P10001-44/26/45 to P4751-2/6/1
	RS 232 CH 2 IN/OUT GMC710	1P10001-47/46/30 to P7101-1/2/6
	ARINC 429 CH 1 IN HIGH SPEED GRS77 #1	1P10001-18/19 to 1P771-13/28
	ARINC 429 CH 2 IN LOW SPEED GDC74 #1	1P10001-16/17 to 1P741-32/33
PFD2	ARINC 429 CH 1 IN HIGH SPEED GRS77 #2	2P10001-18/19 to 2P771-13/28
	ARINC 429 CH 2 IN LOW SPEED GDC74 #2	2P10001-16/17 to 2P741-32/33
	RS 232 CH 1 IN/OUT GCU 476	3P10001-45/44/26/45 to P4751-3/4/8/3
MFD	RS 232 CH 2 IN/OUT GMC710	3P10001-47/46/30/47 to P7101-3/4/8/3
	ARINC 429 CH 1 IN HIGH SPEED GRS77 #1	3P10001-18/19 to 1P771-33/19
	ARINC 429 CH 2 IN LOW SPEED GDC74 #1	3P10001-16/17 to 1P741-41/42
	RS 232 CH 1 IN/OUT GDC74 #1	1P603-41/42/43 to 1P741-11/12/10
GIA 1	RS 232 CH 2 OUT GRS77 #2	1P603-46/45 to 2P771-6/35
	RS 232 CH 3 IN GAE43 ALT ENCODER	1P603-47/48 to P727-1/5
	RS 232 CH 5 IN/OUT GTX33 #1 w/ TIS	1P603-53/55 to 1P3301-23/22
	RS 232 CH 6 IN/OUT GRS77 #1	1P603-56/57/58 to 1P771-26/41/11
	RS 232 CH 7 IN/OUT GMA347 #1	1P603-59/62 to 1P3472-6/7
	ARINC 429 CH 5 IN LOW SPEED GDC74 #1	1P603-12/13 to 1P741-26/27
	ARINC 429 CH 6 IN HIGH SPEED GRS77 #1	1P603-14/15 to 1P771-14/29
	RS 232 CH 1 IN/OUT GDC74 #2	2P603-41/42/43 to 2P741-11/12/10
GIA 2	RS 232 CH 2 OUT GRS77 #1	2P603-46/45 to 1P771-6/35
	RS 232 CH 3 IN GAE43 ALT ENCODER	2P603-47/48 to P727-2/5
	RS 232 CH 4 IN/OUT WX-500	2P603-52/50/5147/48 to P3-8/20 P2-5/8
	RS 232 CH 5 IN/OUT GTX33 #2 (GTX33 #1 if GTX33 #2 option not installed)	2P603-55/53/54 to 2P3301-22/23/51 (1P3301-24/25/58 if GTX33 #2 option not installed)
	RS 232 CH 6 IN/OUT GRS77 #2	2P603-56/57/58 to 2P771-26/41/11
	ARINC 429 CH 4 IN HIGH SPEED TRAFFIC ADVISORY (KTA810)	2P603-10/11 to P10-54/55
	ARINC 429 CH 5 IN LOW SPEED GDC74 #2	2P603-12/13 to 2P741-26/27
	ARINC 429 CH 6 IN HIGH SPEEDGRS77 #2	2P603-14/15 to 2P771-14/29
	ARINC 429 CH 1 OUT LOW SPEED GEN PURPOSE (KTA810)	2P603-70/71/70 to P10-60/61
	ARINC 429 CH 3 OUT HIGH SPEED KTA810	2P603-75/74 to P10-62/63
	RS 485 CH 1 IN/OUT GEA 1	1P603-23/24 to P701-5/6
GIA 1	RS 485 CH 4 IN/OUT GFC700	1P603-4/6 to P832-J/T & P834-J/T 1P603-5/7 to P831-J/T & P833-J/T

CHART 12
NOT USED

PAGES
34250149 THRU 34250159
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E. Recommended Tools

The following tools (or equivalent) are recommended to perform various maintenance tasks on the G1000 system.

(1) Calibrated voltmeter capable of measuring 0-32 Volts DC.
(2) #2 Phillips Screwdriver.
(3) 3/32nd inch Hex Tool.
(4) Calibrated digital Level with 0.25 degrees of accuracy capability.
(5) Calibrated VHF NAV/COM/ILS ramp tester.
(6) Calibrated transponder ramp tester including Mode S capability for Mode S transponder equipped aircraft.
(7) Air Data Test Set (ADTS) capable of simulating altitude up to the aircraft's service ceiling.
(8) Headset/Microphone.
(9) Outdoor line-of-site to GPS satellite signals or GPS indoor repeater.
(10) Ground Power Unit (Capable of supplying 28 Vdc).
(11) Calibrated Flight Control Cable Tension Meter or equivalent.
(12) Servo Mount Slip Clutch Adjustment Tool – Garmin P/N T10-00110-01 and a 2 Amp, 24 V, DC Power Supply.
(13) Calibrated digital thermometer suitable for measuring ambient temperature.
(14) Laptop with RS232 emulation software.
(15) M22885/108T8234 extraction tool.
(16) An 0.060" 6-Spline wrench.
(17) WX-PA portable analyzer kit (If Stormscope is installed).
(18) Calibrated torque wrench capable of measuring 0 – 70 in/lbs.
(19) Standard sockets & wrenches (3/8", 9/16", and 13/16").

F. Inspections

(1) Annual / 100 Hour Inspection

Each 12 months or 100 hours time-in-service, whichever comes first:

(a) See Inspections under Garmin GFC 700 in 22-10-00.

(b) Perform the "G1000 Checks" on page 342501176.
(c) Perform the "Avionics Cooling Fan Fail Annunciation Test" on page 342501174.
(d) For airplanes equipped with GDL 69A, activate the Stall Warning horn and verify that the XM audio is muted.

(2) 500 Hour Inspection

Perform a functional test of the PFD1, MFD, PFD2, and Avionics cooling fans as follows: (see 21-20-00 for cooling fan locations)

(a) Turn OFF all electrical components and avionics systems.

(b) Verify the master switch is ON.

(c) Pull the three or four (as equipped) avionics FAN circuit breakers. I.E., PFD1 FAN, PFD2 FAN, AV/MFD FAN, AV/PFD2 FAN, MFD FAN, and/or AV FANS.

(d) One at a time, push IN each FAN circuit breakers. Verify airflow from each fan by motor sound and pull OUT that circuit breaker before moving to the next.

(e) After checking each fan, turn OFF master switch and push IN all three FAN circuit breakers.

(3) 1000 Hour Inspection

Each 1000 hours time-in-service visually inspect each unit listed below for corrosion, damage, or other defects. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

(a) GDU 1040 PFD and GDU 1500 or GDU 1240A MFD

Remove the PFDs and MFD. Inspect the mounting surfaces and connectors for corrosion, heavy oxidation, or other damage.

(b) PFD, MFD, and Avionics Cooling Fans

Inspect PFD1, PFD2, MFD, and the Avionics cooling fans for accumulation of dust or other damage. Remove excess dust as required. Perform the "500 Hour Inspection".

(c) GMU 44 Magnetometer

Remove the GMU 44 and inspect the mounting hardware and GMU 44 for corrosion or other damage.

(d) For the following components, inspect the individual unit(s), rack(s), and connectors for corrosion or other defects.

1) GMA 347 or GMA 350 Audio Panel

2) GIA 63W Integrated Avionics Units (2 ea.)

3) GEA 71 Engine/Airframe Unit

4) GTX 33/33D (ES) Mode S Transponders (if installed)

5) GDC 74A ADC

6) GRS 77 AHRS

7) GDL 69/69A - Datalink (if installed)

8) GSR 56 Iridium Satellite Transceiver (if installed)

9) GTS 825 TAS (if installed)

(e) Perform G1000 Equipment Electrical Bonding Check in 51-80-00.

(f) Perform visual checks on the shield terminations for any degradation.

G. Components

(1) General

Maintenance of the G1000 system components is "On Condition" only. The following provides basic information regarding unit removal, installation, and troubleshooting. Basic guidance on loading and configuring software is provided under "Software Load" on page 342501152. When removing and/or replacing any G1000 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G1000 LRU, it is required that the technician verify the LRU software part numbers.

To check an LRU software part number and/or version:

(a) Start the G1000 system in configuration mode as described in "Configuration Mode Overview" on page 342501139.

(b) The System Status page shows a list of LRUs in the LRU window. Activate the cursor and highlight the LRU window.

(c) Use the FMS knob to scroll through the list in the window and select the desired LRU.

(d) The software part number and version is displayed in the DATA window. Compare this to the latest configuration shown in the appropriate Required Equipment List (REL), "Chart 39" on page 342501229.

NOTE: If a faulty LRU is not reporting its software version and part number, check aircraft maintenance logs for last software version loaded and verify against the latest configuration.

PFD and MFD Displays
Figure 15

(2) GDU 1040 PFD and GDU 1500 or GDU 1240A MFD

See "Figure 15".

CAUTION: THE GDUS USE A LENS COATED WITH A SPECIAL ANTI-REFLECTIVE COATING THAT IS VERY SENSITIVE TO SKIN OILS, WAXES AND ABRASIVE CLEANERS. CLEANERS CONTAINING AMMONIA WILL HARM THE ANTI-REFLECTIVE COATING. IT IS VERY IMPORTANT TO CLEAN THE LENS USING A CLEAN, LINT-FREE CLOTH AND AN EYEGLASS LENS CLEANER THAT IS SPECIFIED AS SAFE FOR ANTI-REFLECTIVE COATINGS.

(a) Description

Two Garmin GDU 1040 displays and one GDU 1500 or GDU 1240A display are installed in the instrument panel. The GDU 1040 units, 10.4 inch LCD displays with 1024x768 resolution, are configured as PFD1 and PFD2; the GDU 1500 unit, a 15 inch LCD display with 1024x768 resolution (or a GDU 1240A unit, a 12 inch LCD display with 1080x800 resolution), is configured as a MFD. The PFD displays are located on either side of the MFD, with the stand-by instruments located to the left of the Pilot's PFD (PFD1). A GMC 710 AFCS Controller is located in the center of the instrument panel, above or below the MFD. Additionally, a GMA 347 or GMA 350 Audio Panel is located below the MFD, and a GCU 476 is installed in the pedestal. The GCU 476 provides the control interface for the MFD.

The GDU 1500/GDU 1240A communicates with the GDU 1040 units and the GDL 69A (and/or GDL 88) through a high-speed data bus (HSDB) Ethernet connection. The GDU 1500/GDU 1240A communicates with the GCU 476 via RS-232 digital interface.

The GDU 1040 units communicate with each other and the GIA 63W units through a High-Speed Data Bus (HSDB) Ethernet connection.

(b) Troubleshooting

See "Chart 9" on page 34250145, "Chart 13" on page 34250164, and "Chart 14" on page 34250166.

CHART 13 (Sheet 1 of 2)
G1000 - TROUBLESHOOTING - GDU

Symptom	Recommended Action
Display is blank Display resets Display flickers	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power. ✓ If GDU recovers, observe display for yellow text containing error information at the top of the screen. If message indicates software need to be re-loaded, then re-load software. Otherwise, replace the GDU. • Use a bright light to verify LCD is active. ✓ Adjust avionics dimmer control full clockwise. ✓ Manually turn up backlight on the PFD and load configuration files to the GDU. • Ensure slide lock is fully engaged with the locking tabs on the back of the unit. ✓ If slide lock is not fully engaged, remove connector and verify the locking tabs on the GDU are perpendicular to the connector. If necessary, straighten them before reseating connector. • Ensure GDU is receiving power. If a circuit breaker is tripped, determine source of short before resetting breaker. • Ensure circuit breakers have not failed and power wire connections are secure. • Swap PFD1 and PFD2. ✓ If problem follows unit, replace the display. Please note the position it failed in (PFD1/2). ✓ If problem does not follow unit, troubleshoot aircraft wiring for fault.
SD card is stuck in GDU	• DO NOT insert a screwdriver of any length into the card slot. • DO NOT pry against the overlay. • DO NOT force the SD Card out. • Use a small screwdriver in the groove on the side of the exposed end of the card to help pull out the card. • Push the card in further to release the card locking mechanism. • Check SD Card for having more than one label. Two or more labels on the card will cause sticking. ✓ Remove all but one label. • Ensure the SD card is from SanDisk. Use of other SD Cards is not recommended. • If card was inserted with the label facing to the right, do not attempt to remove. Return the unit to Garmin for repair.
A button/knob/joystick does not appear to function	• Go to the GDU TEST page in configuration mode and verify button, knob, or joystick operates correctly by observing a change in color from red to green in the button/knob/joystick icon when the button/knob/joystick is pressed. If a button is stuck, the button icon will be green without pressing the button as soon as you turn to the GDU TEST page. ✓ If problem is verified, replace GDU.

CHART 13 (Sheet 2 of 2)
G1000 - TROUBLESHOOTING - GDU

<table>
<tr><th>Symptom</th><th>Recommended Action</th></tr>
<tr><td>Terrain/Obstacle/Safetaxi does not display</td><td>• Ensure supplemental data cards are inserted correctly in the lower slots of all three GDU’s.
• Allow the system to verify the data on the cards for approximately five minutes after power-up.
• If a database does not activate, reload the problem database onto the SD Card or replace the card. Replacing all data cards in a set will keep database cycles the same and should prevent database mismatch errors.</td></tr>
<tr><td>Display will not track dimmer bus</td><td rowspan="2">• Reload GDU configuration files.
• If display is a PFD, swap PFD1 and PFD2 to see if problem remains with display.
✓ Replace display if condition remains with the same unit.
✓ If condition remains in original position after swapping displays, check GDU dimmer input to verify voltage is present.
• If display is the MFD, check dimmer input to verify voltage is present.
✓ Replace MFD if dimmer voltage is present.</td></tr>
<tr><td>Keyboard will not track dimmer bus</td></tr>
</table>

CHART 14 (Sheet 1 of 8)
G1000 - MESSAGE ADVISORIES - GDU

Failure Message	Cause	Solution
SW MISMATCH – GDU software version mismatch. Xtalk is off.	The system has found the PFDs and/or MFD software versions do not match.	• Load correct software version. See Software Loading procedure.
MANIFEST – PFD 1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in the specified PFD.	
MANIFEST – PFD 2 software mismatch. Communication Halted.		
MANIFEST – MFD1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in MFD.	
CNFG MODULE – PFD 1 configuration module is inoperative.	The PFD master configuration module has failed.	• Check master configuration module connector and wiring for damage inside the GDU connector backshell. ✓ Replace master configuration module wiring and pins. ✓ If problem persists, replace master configuration module. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

CHART 14 (Sheet 2 of 8)
G1000 - MESSAGE ADVISORIES - GDU

<table>
<tr><th>Failure Message</th><th>Cause</th><th>Solution</th></tr>
<tr><td>MFD1 CONFIG – MFD1 configuration error. Config service req’d.</td><td rowspan="3">A configuration mismatch has occurred between the display and the Master Configuration Module.</td><td rowspan="3">• Reload the display configuration files from SD Loader Card.
• Reload system configuration files by pressing the UPDT CFG softkey on the Configuration Upload Page in the PFD1 System Page Group to load configuration files into the configuration module.
✓ If message persists, check PFD1 config module wiring for faults and replace if necessary.
✓ If issue continues, replace PFD1 master configuration module.
✓ If problem persists, replace the display.

NOTE
New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.</td></tr>
<tr><td>PFD 1 CONFIG – PFD 1 configuration error. Config service req’d.</td></tr>
<tr><td>PFD 2 CONFIG – PFD 2 configuration error. Config service req’d.</td></tr>
</table>

CHART 14 (Sheet 3 of 8)
G1000 - MESSAGE ADVISORIES - GDU

Database Message Advisories		
Failure Message	**Cause**	**Solution**
MFD1 DB ERR – MFD1 navigation database error exists. PFD1 DB ERR – PFD1 navigation database error exists. PFD2 DB ERR – PFD2 navigation database error exists.	The MFD or specified PFD has encountered an error in the Jeppesen navigation database.	Reload navigation database into the display. Contact Garmin Technical Support for assistance.
MFD1 DB ERR – MFD1 basemap database error exists.	The MFD has encountered an error in the basemap database.	Confirm supplemental data card is inserted fully in the bottom of the display. Move the data card to the top slot of the display. If the error clears, the problem is with the bottom slot. Insert and remove a SD card multiple times to clean the contacts. If the card still does not work in the bottom slot, leave the card in the top slot or replace the display. Swap with a supplemental data card from another display in the same system. For basemap only, if problem remains in the same GDU, contact Garmin Aviation Product Support to see if a Basemap file may be obtained to load into the display. For all others, if problem remains in the same GDU, replace that GDU If problem moves to the other display, replace the supplemental data card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.
PFD1 DB ERR – PFD1 basemap database error exists. PFD2 DB ERR – PFD2 basemap database error exists.	The specified PFD has encountered an error in the basemap database.	
MFD1 DB ERR – MFD1 terrain database error exists.	The MFD has encountered an error in the terrain database.	
PFD1 DB ERR – PFD1 terrain database error exists. PFD2 DB ERR – PFD2 terrain database error exists.	The specified PFD has encountered an error in the terrain database.	
MFD1 DB ERR – MFD1 obstacle database error exists.	The MFD has encountered an error in the obstacle database.	
PFD1 DB ERR – PFD1 obstacle database error exists. PFD2 DB ERR – PFD2 obstacle database error exists.	The specified PFD has encountered an error in the obstacle database.	
MFD1 DB ERR – MFD1 airport terrain database error exists.	The MFD has encountered an error in the airport terrain database.	
PFD1 DB ERR – PFD1 airport terrain data base error exists. PFD2 DB ERR – PFD2 airport terrain data base error exists.	The specified PFD has encountered an error in the airport terrain database.	

CHART 14 (Sheet 4 of 8)
G1000 - MESSAGE ADVISORIES - GDU

Database Message Advisories (cont.)		
Failure Message	**Cause**	**Solution**
MFD1 DB ERR – MFD1 SafeTaxi database error exists.	The MFD has encountered an error in the SafeTaxi database.	Confirm supplemental data card is inserted fully in the bottom of the display. Move the data card to the top slot of the display. If the error clears, the problem is with the bottom slot. Insert and remove a SD card multiple times to clean the contacts. If the card still does not work in the bottom slot, leave the card in the top slot or replace the display. Swap with a supplemental data card from another display in the same system. If problem remains in the same GDU, replace that GDU. If problem moves to the other display, replace the supplemental data card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.
PFD1 DB ERR – PFD1 SafeTaxi data base error exists. PFD2 DB ERR – PFD2 SafeTaxi data base error exists.	The specified PFD has encountered an error in the SafeTaxi database.	
MFD1 DB ERR – MFD1 terrain database missing. PFD1 DB ERR – PFD1 terrain database emissing. PFD2 DB ERR – PFD2 terrain database missing.	The terrain database is present on another LRU, but is missing on the specified LRU.	Ensure data card is properly inserted. Swap with a supplemental data card from another display in the same system. If problem remains in the same GDU, replace that GDU. If problem moves with the data card, replace card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.

CHART 14 (Sheet 5 of 8)
G1000 - MESSAGE ADVISORIES - GDU

Database Message Advisories (cont.)

Failure Message	Cause	Solution
MFD1 DB ERR – MFD1 obstacle database missing. PFD1 DB ERR – PFD1 obstacle database missing. PFD2 DB ERR – PFD2 obstacle database missing.	The obstacle database is present on another LRU, but is missing on the specified LRU.	Ensure data card is properly inserted. Swap with a supplemental data card from another display in the same system. If problem remains in the same GDU, replace that GDU. If problem moves with the data card, replace card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.
MFD1 DB ERR – MFD1 airport terrain database is missing. PFD1 DB ERR – PFD1 airport terrain database is missing. PFD2 DB ERR – PFD2 airport terrain database is missing.	The airport terrain database is present on another LRU, but is missing on the specified LRU.	
MFD DB ERR – MFD ChartView database error.	The MFD has encountered an error in the ChartView database.	Confirm supplemental data card is inserted fully in the bottom of the display. Move the data card to the top slot of the display. If the error clears, the problem is with the bottom slot. Insert and remove a SD card multiple times to clean the contacts. If the card still does not work in the bottom slot, leave the card in the top slot or replace the display. Reload Chartview, FlightCharts, or Airport Directory, as appropriate. Replace the supplemental data card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors. Replace MFD.
MFD DB ERR – MFD FliteCharts database error exists.	The MFD has encountered an error in the FliteCharts database.	
MFD1 DB ERR - MFD1 Airport Directory database error exists. PFD1 DB ERR - PFD1 Airport Directory database error exists.	The LRU has encountered an error in the Airport Directory Database.	

CHART 14 (Sheet 6 of 8)
G1000 - MESSAGE ADVISORIES - GDU

Database Message Advisories (cont.)		
Failure Message	**Cause**	**Solution**
DB MISMATCH – Navigation database mismatch. Xtalk is off.	The G1000 has found the Jeppesen navigation database in the PFDs and MFD do not match.	Load the same cycle of navigation database to all displays.
DB MISMATCH – Standby navigation database mismatch.	The G1000 has found the standby Jeppesen navigation database in the PFDs and MFD do not match.	Load the same cycle of standby navigation database to all displays.
DB MISMATCH – Terrain database mismatch.	The PFD and/or MFD have different terrain database versions installed.	Confirm supplemental data card is inserted fully. Reload the database in each LRU and/or replace the supplemental datacard(s) in each LRU.
DB MISMATCH – Obstacle database mismatch.	The PFD and/or MFD have different obstacle database versions installed.	
DB MISMATCH – Airport Terrain database mismatch.	The PFD and/or MFD have different airport terrain database versions installed.	

Cooling Message Advisories		
Failure Message	**Cause**	**Solution**
MFD1 COOLING – has poor cooling. Reducing power usage.	MFD1 has exceeded its operating temperature range.	Check cooling fan and wiring for proper operation. Replace cooling fan if unable to determine if operating correctly. If problem persists, replace the affected LRU. If problem continues contact Garmin Aviation Product Support for assistance.
PFD1 COOLING – has poor cooling. Reducing power usage.	PFD1 has exceeded it’s operating temperature range.	
PFD2 COOLING – has poor cooling. Reducing power usage.	PFD2 has exceeded it’s operating temperature range.	

CHART 14 (Sheet 7 of 8)
G1000 - MESSAGE ADVISORIES - GDU

Key Message Advisories		
Failure Message	**Cause**	**Solution**
MFD1 “key” KEYSTK – key is stuck.	The SYSTEM has determined a key is stuck on MFD1.	• Go to the GDU TEST page in configuration mode and verify key is stuck (if key is stuck the corresponding indicator will be green). • Exercise suspected stuck key and reset GDU TEST page to see if indicator remains green without pressing the key. ✓ If problem persists replace the display.
PFD 1 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 1.	
PFD 2 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 2.	

Miscellaneous Message Advisories		
Failure Message	**Cause**	**Solution**
XTALK ERROR – A flight display cross talk error has occurred.	A communication error has occurred between the MFD and/or PFDs.	• Ensure a database error has not occurred (identified in the ALERTS window on the PFD). ✓ If a database error has occurred, correct error before proceeding. • Check display Ethernet interconnect wiring. • Replace PFD1 with a known good unit, to verify location of problem: ✓ If problem persists, reinstall original PFD1 and replace PFD2. ✓ If problem persists, reinstall PFD2 and replace MFD.
DATA LOST – Pilot stored data lost. Recheck settings.	Pilot stored data has been lost.	• If the CLR key was held during a power cycle, disregard message. • Cycle power to PFD1: ✓ Ensure CLR key is not stuck on the GDU TEST page. ✓ If problem persists, replace PFD1.

CHART 14 (Sheet 8 of 8)
G1000 - MESSAGE ADVISORIES - GDU

Miscellaneous Message Advisories		
Failure Message	**Cause**	**Solution**
MFD1 SERVICE – needs service. Return unit for repair.	The G1000 has determined MFD1 needs service.	• Ensure the MFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the MFD • Replace the MFD
PFD 1 SERVICE – needs service. Return unit for repair.	The G1000 has determined the specified PFD needs service.	• Ensure the PFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the PFD • Replace the PFD.
PFD 2 SERVICE – needs service. Return unit for repair.		
PFD 1 VOLTAGE – PFD 1 has low voltage. Reducing power usage.	The specified PFD supply voltage is low.	• Check input voltage to PFD. • If input voltage is ok, replace PFD..
PFD 2 VOLTAGE – PFD 2 has low voltage. Reducing power usage.		
MFD1 VOLTAGE – MDF1 has low voltage. Reducing power usage.	The MFD supply voltage is low.	• Check input voltage to MFD. • If input voltage is ok, replace MFD.

(c) Removal

1) Using a 3/32nd hex tool, rotate all four ¼-turn fasteners counter-clockwise until they reach their stops.

2) Carefully remove the display from the panel.

3) While supporting the display, disconnect the connector.

(d) Installation

1) Visually inspect the connector and pins for signs of damage. Repair any damage. While supporting the display, connect the connector to the rear of the unit.

2) Carefully insert the display into the panel cutout, ensuring that all 4 ¼-turn fasteners align with the corresponding holes.

3) Seat the display in the panel cutout. Do not use excessive force while inserting the display.

4) Once seated, rotate all four ¼-turn fasteners clockwise to lock the display to the panel.

(e) Return to Service

1) Original Display Reinstalled.

If the removed display(s) are re-installed in their original positions, no software or configuration loading is required. Continue to the "PFD/MFD Test" on page 342501181.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original PFD Displays Installed in Opposite Locations for Troubleshooting.

If the PFD1 and PFD2 are installed in opposite positions, no software or configuration loading is required. Continue to the "PFD/MFD Test" on page 342501181.

3) New, Repair, or Exchanged Display(s) Installed.

If a new, repaired or exchanged GDU 1040, GDU 1500, or GDU 1240A is installed, the correct software and configuration files must be loaded to the unit. See "G1000 Software/Configuration Procedure" on page 342501150. If ChartView or TAWS were previously installed, these must be reactivated. See Chartview and TAWS Unlock Procedure respectively. If any other options were previously installed, these must also be reactivated per "Optional Equipment Configuration" on page 342501154. Then continue to the "PFD/MFD Test" on page 342501181.

(3) GCU 476 PFD/MFD Control Unit

(a) Description

This pedestal mounted keypad provides MFD/PFD and radio tuning control through an RS-232 digital interface. See "Figure 1" on page 3425011 and "Figure 7" on page 34250112.

(b) Troubleshooting

See "Chart 15" on page 34250176.

(c) Removal

1) Turn each of the four locking sockets one quarter turn counterclockwise until they reach their stops using a 3/32” hex drive tool.

2) Pull unit up gently to remove unit from the panel.

3) Remove connector from the back of the unit by pushing on the slide lock tab on the side of the connector to release it and then pull connector away from unit. If unit connector uses thumb screws instead of a slide lock, they must be disconnected on both sides.

(d) Installation

1) Inspect wire harness connector for damaged pins before installing the new unit.

2) Attach connector to the back of the unit by first pressing the slide lock tab on the side of the connector, then align and mate the unit and wire harness connectors together, and then release the slide lock tab. Locking tabs must be engaged on both ends of the connector. If the unit connector uses thumb screws instead of a slide lock, they must be tightened on both sides securely.

3) Install and hold the unit flush with the instrument panel and ensure that locking stud alignment marks are in the vertical position.

4) Turn each of the four locking sockets one quarter turn clockwise using a 3/32” hex drive tool (this may require applying a small amount of forward pressure to engage the quarter turn sockets).

(e) Return to Service

1) Original GCU 476 Reinstalled

No software or configuration loading is required if the removed GCU 476 is re-installed. Continue to the "GCU 476 Test" on page 342501199.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GCU 476 Installed

If a new, repaired, or exchanged GCU 476 is installed, the correct software and configuration files must be loaded to the unit. See "G1000 Software/ Configuration Procedure" on page 342501150 and then continue to the "GCU 476 Test" on page 342501199.

CHART 15
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GCU 476

Failure Messages	Cause	Solutions
GCU CNFG – GCU Config error. Config service req'd.	The G1000 has detected a GCU 476 configuration	• Load GCU configuration files. ✓ Replace GCU. ✓ If problem persists, replace master configuration module[1], check config module wiring for faults and replace if necessary.
GCU FAIL – GCU is inoperative.	The G1000 has detected a failure in the GCU 476.	• Check wiring for faults per "Failed Path Messages." • Replace the GCU 476.
MANIFEST – GCU software	The system has detected an incorrect software version loaded in the GCU 476.	• Load the correct software version. See G1000 Software/Configuration Procedures, below.
GCU KEYSTK – GCU [key name] key is stuck.	A key is stuck on the GCU bezel.	Exercise the key to free it. If the problem persists, replace the GCU 476.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

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(4) GMA 347 Audio Panel

S/N's 4636460, 4636463–4636651, less 4636481 and 4636633; and 4692134 and up, less 4692141, 4692149, and 4692153.

See "Figure 16".

(a) Description

The Garmin GMA 347 Audio Panel integrates NAV/COM digital audio, intercom system and marker beacon controls. The GMA 347 panel provides control of all cockpit intercom/mic systems as well as NAV/COM/ILS audio. The unit also provides display reversion mode control through a large red Display button. Power to the audio panel is provided by the battery switch or by applying external power. The GMA 347 unit interfaces with the existing marker beacon antenna as well as existing mic and phone jacks.

(b) Troubleshooting

See "Chart 16" on page 34250180 and "Chart 17" on page 34250182.

(c) Removal

1) Using a 3/32nd hex tool, turn the hex nut counter-clockwise until the GMA 347 is unlocked from its location.

2) Carefully remove the GMA 347 from its rack.

(d) Installation

1) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.

2) Gently insert the GMA 347 into the rack until the locking tab engages the rack.

3) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not overtighten the nut.

(e) Return to Service

1) Original GMA 347 Reinstalled.

No software/configuration loading or testing is required if the removed GMA 347 is reinstalled. Continue to the final "Return to Service Procedure" on page 342501213.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GMA 347 Installed.

If a new, repaired, or exchanged GMA 347 is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 Software/Configuration Procedure" on page 342501150 and then continue to the "GMA 347 Test" on page 342501181.

GMA 347 Audio Panel
Figure 16

(5) GMA 350 Audio Panel

S/N's 4636633, 4636652 and up.

See "Figure 17".

(a) Description

The Garmin GMA 350 Audio Panel integrates NAV/COM digital audio, intercom system and marker beacon controls. The GMA 350 panel provides control of all cockpit intercom/ mic systems as well as NAV/COM/ILS audio. Power to the audio panel is provided by the battery switch or by applying external power. The GMA 350 unit interfaces with the marker beacon antenna as well as mic and phone jacks and oxygen mask mic.

(b) Troubleshooting

See "Chart 16" on page 34250180 and "Chart 17" on page 34250182.

(c) Removal

1) Using a 3/32nd hex tool, turn the hex nut counter-clockwise until the GMA 350 is unlocked from its location.

2) Carefully remove the GMA 350 from its rack.

(d) Installation

1) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.

2) Gently insert the GMA 350 into the rack until the locking tab engages the rack.

3) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not overtighten the nut.

(e) Return to Service

1) Original GMA 350 Reinstalled.

No software/configuration loading or testing is required if the removed GMA 350 is reinstalled. Continue to the "GMA 350 Test" on page 342501182.

NOTE: This does not include units that were returned for repair (see below) as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GMA 350 Installed.

If a new, repaired, or exchanged GMA 350 is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 Software/Configuration Procedure" on page 342501150, and then continue to the "GMA 350 Test" on page 342501182.

GMA 350 Audio Panel
Figure 17

CHART 16
G1000 - TROUBLESHOOTING - GMA

Symptom/Failure Message	Recommended Action
Noise in Audio	Most often the cause of the noise is external to the GMA. Try the following to locate the source of the noise before replacing the GMA: • Try a different pair of headsets. Noise cancelling headsets may pick up and/or generate more noise than standard headsets from their own circuitry. • Check for noise with the engine turned off. ✓ If the noise is present only when the engine is running, check the generator and/or ignition system as possible sources of noise (see applicable airframe specific maintenance manual). • Check for noise as all electrical equipment is turned on and off (strobes, other radios, etc.). ✓ If the noise is identified from one electrical system or component refer to the applicable airframe specific maintenance manual. • Ensure the NAV/COM squelch is not open. • Ensure the ADF and DME audio is not active. • Ensure the marker beacon audio is not active. • Ensure the ICS squelch is not open. ✓ Master squelch level can be adjusted on the GMA CONFIGURATION page for higher noise environments. • Replace unit only after all possible external sources of noise are eliminated.
Buttons Do Not Work.	• Some buttons are disabled in the GMA CONFIGURATION page by default. This is to remove potential sources of audio noise for inputs that are not used. If in doubt as to which buttons should be disabled, reload GMA config files and other config files for optional equipment installed in the aircraft (i.e. ADF, HF, etc.) from the loader card.
Speaker Cuts Out	• Reduce volume level of the item that caused the speaker to cut out when turned up. A speaker protection circuit disables the speaker output if the volume is too high. If the volume is not sufficient, replace aircraft cabin speaker.
Mic Audio Heard in Speaker	• Reduce ICS Volume.

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CHART 17 (Sheet 1 of 2)
G1000 - MESSAGE ADVISORIES - GMA

Failure Message	Cause	Solution
MANIFEST – GMA1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GMA #1.	Load correct software version.
MANIFEST – GMA2 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GMA #2.	Load correct software version.
GMA1 SERVICE – GMA1 needs service. Return unit for repair.	The G1000 has determined GMA#1 needs service.	Replace GMA #1.
GMA2 SERVICE – GMA2 needs service. Return unit for repair.	The G1000 has determined GMA#2 needs service.	Replace GMA #2.
GMA1 FAIL – GMA 1 is inoperative.	The G1000 has detected a failure in GMA#1.	Ensure GMA #1, both GIAs and all GDUs are receiving power. Troubleshoot for a failed datapath per "Failed Path Messages." Replace GMA #1.
GMA2 FAIL – GMA 2 is inoperative.	The G1000 has detected a failure in GMA#2.	Ensure GMA #2, both GIAs and all GDUs are receiving power. Troubleshoot for a failed datapath per "Failed Path Messages." Replace GMA #2.
GMA1 CONFIG – GMA1 configuration error. Config service req'd.	The G1000 has detected a GMA #1 configuration mismatch.	Reload affected GMA Configuration files including optional equipment configuration files that list that GMA in the PRODUCT box on the System Upload page. Replace affected GMA. If problem persists, replace master configuration module, check config module wiring for faults and replace if necessary.
GMA2 CONFIG – GMA2 configuration error. Config service req'd.	The G1000 has detected a GMA #2 configuration mismatch.	

NOTE: New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

CHART 17 (Sheet 2 of 2)
G1000 - MESSAGE ADVISORIES - GMA

Failure Message	Cause	Solution
GMA XTALK – GMA crosstalk error has occurred.	The G1000 has detected an error in the communication between GMA #1 and GMA #2.	Ensure both units are receiving power. Ensure both units are configured. Check interconnect wiring and connector pins for faults. Replace GMA #1 with a known good unit. If problem persists, reinstall original GMA #1 and replace GMA #2.
BACKUP PATH – Audio panel using backup data path.	The GMA is using a backup RS-232 data path.	Troubleshoot for a failed per “Failed Path Messages.”

(6) GIA 63W Integrated Avionics Unit

See "Figure 18".

(a) Description

Two Garmin GIA 63W Integrated Avionics Units (IAUs) contain the VHF COM/NAV receivers, WAAS GPS receiver, Flight Director, and system integration microprocessors. The GIAs also serve as a communication interface to all other G1000 LRUs in the system. Each GIA 63W communicates directly with the on-side GDU 1040 display using a HSDB Ethernet connection. Both GIAs are located directly behind the PFD1 and PFD2, installed into their respective LRU racks. Power is provided to GIA1 by the main bus or the emergency bus. GIA2 is powered by #2 avionics bus.

GIA 63W Integrated Avionics Unit
Figure 18

1) Both GIA 63s interface to the following equipment:
- VOR/LOC/Glideslope Antenna System
- VHF COM #1 & #2 Antennas
- GA 35 GPS/WAAS Antennas
- GMA 347 or GMA 350
- GEA 71,
- GDU 1040, #1 & #2
- GSA 8X Servo Actuators (all)
- GRS 77, #1 & #2

2) The GIA 63 #1 interfaces to the following additional equipment:
- GDC 74A #1
- GTX 33 #1
- GTS 825 TAS (if Installed, S/N's 4636633, 4636652 and up)

3) The GIA 63 #2 interfaces to the following additional equipment:
- GDU 1500 or GDU 1240A
- GDC 74A #2
- GTX 33 #2 (if installed)
- ADF (if installed)
- DME (if Installed).
- KTA 810 TAS (if Installed, S/N's 4636460, 4636463–4636651, less 4636481 and 4636633; and 4692134 and up, less 4692141, 4692149, and 4692153)
- Stormscope (if installed)

NOTE: The GIA 63W is compatible only with the antennas listed in the GIA 63W Installation Manual, Garmin P/N 190-00303-05.

(b) Troubleshooting

See "Chart 18" on page 34250186 and "Chart 19" on page 34250188.

(c) Removal

1) Remove the appropriate PFD as described above.

2) Unlock the GIA 63W handle by loosening the Phillips screw on the handle.

3) Pull the handle upward to unlock the GIA 63W. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

2) Gently insert the GIA 63W into its rack. The handle should engage the dogleg track.

3) Press down on the GIA 63W hand to lock the unit into the rack.

4) Lock the handle to the GIA 63W body using the Phillips screw.

5) Reinstall the MFD as described above.

(e) Return to Service

1) Original GIA 63W(s) Reinstalled

No software or configuration loading is required if the removed GIA is reinstalled in its original position (GIA1 and GIA2 in their original racks).

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GIA 63W(s) Swapped for Troubleshooting

No software loading is required if the originally installed GIA units are reinstalled in opposite positions (GIA1 and GIA2 in opposite unit racks). However, configuration loading is required.

See "G1000 Software/Configuration Procedure" on page 342501150, and then continue to the "GIA 63W Test" on page 342501184.

3) New, Repaired, or Exchanged GIA 63W(s) Installed

If a new, repaired, or exchanged GIA 63 is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 Software/Configuration Procedure" on page 342501150, and then continue to the "GIA 63W Test" on page 342501184.

CHART 18 (Sheet 1 of 2)
G1000 - TROUBLESHOOTING - GIA 63W

COM	
Symptom	**Recommended Action**
Weak COM transmit power	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
Weak COM receiver	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
No COM sidetone	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem persists, replace GMA1 with a known good unit. ✓ If problem persists, reinstall original GMA1 and replace GMA2.

NAV	
Symptom	**Recommended Action**
Weak NAV receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

Glideslope	
Symptom	**Recommended Action**
Weak G/S receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

CHART 18 (Sheet 2 of 2)
G1000 - TROUBLESHOOTING - GIA 63W

GPS	
Symptom	**Recommended Action**
Will not acquire satellites.	Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.). Check the aircraft records for the model and serial number of the GPS antenna. If Garmin GA35 antennas below S/N 60000 are installed, one of the antennas may be the source of interference. Replace both GPS antennas to correct. Check date and time on Date/Time Setup Page. If date and time are incorrect, enter the correct date and time. Swap GIA1 and GIA2, reconfigure both GIA's to their new locations, to verify location of problem. If problem follows unit, clear the GPS almanac by performing the following steps: Using the PFD in config mode, go to the GIA RS-232/ARINC 429 Config Page. At the top of the screen, select the GIA that cannot acquire satellites (GIA1 or GIA2) and press the ENT key. Press the "CLR NV" softkey at the bottom of the screen. Select "OK" in the "Clear GIA nonvolatile memory?" pop-up window. Reload GIA Audio and Config files from a loader card. Be sure to reload the config files for any optional equipment installed on the aircraft that require the GIA config to be updated. Cycle power on the system and allow it to restart in normal mode. Place the aircraft outside and allow 15-30 minutes for the GPS to acquire a position and download a new almanac. If clearing nonvolatile memory is unsuccessful and the GPS still cannot acquire a position, replace the GIA. Check GPS antenna and cabling.

CHART 19 (Sheet 1 of 7)
G1000 - MESSAGE ADVISORIES - GIA 63W

COM		
Failure Message	**Cause**	**Solutions**
COM1 SERVICE – COM1 needs service. Return unit for repair.	The system has determined COM1 needs service.	• Replace GIA1.
COM2 SERVICE – COM2 needs service. Return unit for repair.	The system has determined COM2 needs service.	• Replace GIA2.
COM1 PTT – COM1 push-to-talk key is stuck.	The COM1 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA1/GMA #1 interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped; see Software/Configuration above): ✓ If problem follows the unit, replace GIA1. ✓ If problem persists replace defective GMA #1.
COM2 PTT – COM2 push-to-talk key is stuck.	The COM2 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA2/GMA (#2, if installed) interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/Configuration above): ✓ If problem follows the unit, replace GIA2. ✓ If problem persists replace defective GMA (#2, if installed).

CHART 19 (Sheet 2 of 7)
G1000 - MESSAGE ADVISORIES - GIA 63W

<table>
<tr><th colspan="3">COM</th></tr>
<tr><th>Failure Message</th><th>Cause</th><th>Solutions</th></tr>
<tr><td>COM1 RMT XFR – COM1 remote transfer key is stuck.</td><td>The COM1 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td><td>Press the COM1 external remote transfer switch again to cycle its operation.
Check COM1 external remote transfer switch and wiring.
Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above):
✓ If problem follows the unit, replace GIA1.
✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.</td></tr>
<tr><td>COM2 RMT XFR – COM2 remote transfer key is stuck.</td><td>The COM2 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td><td>Press the COM2 external remote transfer switch again to cycle its operation.
Check COM2 external remote transfer switch and wiring.
Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above):
✓ If problem follows the unit, replace GIA2.
✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.</td></tr>
<tr><td>COM1 TEMP – COM1 over temp. Reducing transmitter power.</td><td rowspan="2">The specified COM transceiver is reporting a high temperature condition and is reducing transmit power to prevent damage.</td><td rowspan="2">• Check fan, wiring and air tubing for proper operation (if applicable).
• Replace cooling fan if unable to determine if operating correctly.
• Replace GIA.
• If problem persists contact Garmin Aviation Product Support for assistance.</td></tr>
<tr><td>COM2 TEMP – COM2 over temp. Reducing transmitter power.</td></tr>
</table>

CHART 19 (Sheet 3 of 7)
G1000 - MESSAGE ADVISORIES - GIA 63W

<table>
<tr><th colspan="3">NAV</th></tr>
<tr><th>Failure Message</th><th>Cause</th><th>Solution</th></tr>
<tr><td>NAV1 SERVICE – NAV1 needs service. Return unit for repair.</td><td>The system has detected a failure in NAV1 receiver.</td><td>Replace GIA1.</td></tr>
<tr><td>NAV2 SERVICE – NAV2 needs service. Return unit for repair.</td><td>The system has detected a failure in NAV2 receiver.</td><td>Replace GIA2.</td></tr>
<tr><td>NAV1 RMT XFR – NAV1 remote transfer key is stuck.</td><td>The NAV1 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td><td rowspan="2">Press the affected NAV external remote transfer switch again to cycle its operation.

Check affected NAV remote transfer switch and wiring.

Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above):

If problem follows unit, replace affected GIA.

If problem persists, it’s in the remote transfer switch and/or wiring.</td></tr>
<tr><td>NAV2 RMT XFR – NAV2 remote transfer key is stuck.</td><td>The NAV2 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td></tr>
<tr><td>NAV1 MANIFEST – NAV1 software mismatch. Communication halted.</td><td>The system has detected an incorrect software version loaded in GIA1.</td><td>Load the correct software. See G1000 Software/Configuration Procedure, below.</td></tr>
<tr><td>NAV2 MANIFEST – NAV2 software mismatch. Communication halted.</td><td>The system has detected an incorrect software version loaded in GIA2.</td><td>Load the correct software. See G1000 Software/Configuration Procedure, below.</td></tr>
</table>

CHART 19 (Sheet 4 of 7)
G1000 - MESSAGE ADVISORIES - GIA 63W

Glideslope		
Failure Message	**Cause**	**Solution**
G/S1 SERVICE – G/S1 needs service. Return unit for repair.	The system has detected a failure in G/S1 receiver.	Replace GIA1.
G/S2 SERVICE – G/S2 needs service. Return unit for repair.	The system has detected a failure in G/S2 receiver.	Replace GIA2.
G/S1 FAIL – G/S1 is inoperative.	The system has detected a failure in G/S1 receiver.	Switch GIA1 and GIA2 to verify location of problem: If problem follows the unit, replace affected GIA. If problem does not follow unit, check affected G/S antenna and cabling.
G/S2 FAIL – G/S2 is inoperative.	The system has detected a failure in G/S2 receiver.	

GPS		
Failure Message	**Cause**	**Solution**
NOTE: Before troubleshooting, ensure that no cell phones or devices using cell phone technology are turned on, even in a monitoring state, in the cabin.		
GPS1 SERVICE – GPS1 needs service. Return unit for repair.	The system has detected a failure in GPS1 receiver.	Replace GIA1.
GPS2 SERVICE – GPS2 needs service. Return unit for repair.	The system has detected a failure in GPS2 receiver.	Replace GIA2.
GPS1 FAIL – GPS1 is inoperative.	The system has detected a failure in GPS1 receiver.	Switch GIA1 and GIA2, to verify location of problem: If problem follows the unit, replace affected GIA. If problem does not follow the unit, check affected GPS antenna and cabling.
GPS2 FAIL – GPS2 is inoperative.	The system has detected a failure in GPS2 receiver.	

CHART 19 (Sheet 5 of 7)
G1000 - MESSAGE ADVISORIES - GIA 63W

GPS (cont.)		
Failure Message	**Cause**	**Solution**
LOI – GPS integrity lost. Crosscheck with other NAVS.	If the primary receiver is a WAAS sensor, the alert indicates that GPS position data has timed out.	• Verify the area the aircraft was traveling through did not have loss of GPS coverage. FAA NOTAMs may be issued for periods of outages, or the US Coast Guard website http://www.navcen.uscg.gov/ will have notices posted. • Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.). • If GPS receivers can not aquire a position lock, troubleshoot per the “Will not acquire satellites” section. • Check GPS antenna and cabling.
GPS NAV LOST – Loss of GPS navigation. Insufficient satellites.	There is no GPS position fix available or the system is in dead reckoning mode.	
GPS NAV LOST – Loss of GPS navigation. Position error.	The G1000 has detected an internal position warning has occurred.	
GPS NAV LOST – Loss of GPS navigation. GPS fail.	The G1000 has detected a GPS engine failure.	

Cooling		
Failure Message	**Cause**	**Solution**
GIA1 COOLING – GIA1 temperature too low.	GIA1 operating temperature is too low.	• Allow unit to warm up.
GIA2 COOLING – GIA2 temperature too low.	GIA2 operating temperature is too low.	• Allow unit to warm up.
GIA1 COOLING – GIA1 over temperature.	GIA1 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA1. • If problem persists contact Garmin Aviation Product Support for assistance.
GIA2 COOLING – GIA2 over temperature.	GIA2 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA2. • If problem persists contact Garmin Aviation Product Support for assistance.

CHART 19 (Sheet 6 of 7)
G1000 - MESSAGE ADVISORIES - GIA 63W

Configuration		
Failure Message	**Cause**	**Solution**
MANIFEST – GIA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA1.	Load the correct G1000 system software. See G1000 Software/Configuration Procedure, below.
MANIFEST – GIA2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA2.	
MANIFEST – GFC software mismatch. Communication halted.	Incorrect servo software installed or gain settings are incorrect.	
COM1 CONFIG – COM1 configuration error. COM2 CONFIG – COM2 configuration error.	COM1 and/or COM2 configuration settings do not match backup configuration memory.	
COM1 MANIFEST – COM1 software mismatch. COM2 MANIFEST – COM2 software mismatch.	COM1 and/or COM2 software mismatch, communication halted.	
GIA1 CONFIG – GIA1 audio config error. Config service req'd. GIA2 CONFIG – GIA2 audio config error. Config service req'd.	The GIA's audio configuration files are incorrect or missing.	Reload GIA Audio software and configuration files. Replace master configuration module(1), check config module harness for faults and replace if necessary.
GIA1 CONFIG – GIA1 configuration error. Config service req'd. GIA2 CONFIG – GIA2 configuration error. Config service req'd.	The system has detected a GIA configuration mismatch. If GIAs are not properly configured after being swapped/replaced, this message appears.	Load the configuration files for that GIA. See G1000 Software/ Configuration Procedure, below. If problem persists, replace master configuration module(1), check config module harness for faults and replace if necessary.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

CHART 19 (Sheet 7 of 7)
G1000 - MESSAGE ADVISORIES - GIA 63W

Configuration (cont.)		
Failure Message	**Cause**	**Solution**
HW MISMATCH – GIA hardware mismatch, GIA1 communication halted. HW MISMATCH – GIA hardware mismatch, GIA2 communication halted.	The G1000 has detected a non-WAAS GIA 63.	Replace GIA with a WAAS unit.
GIA1 SERVICE – GIA1 needs service. Return unit for repair. GIA2 SERVICE – GIA2 needs service. Return unit for repair.	The G1000 has detected a failure in the specified GIA.	Replace suspect GIA.

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(7) GAE 43 Cabin Altitude Encoder

See "Figure 19".

(a) Description

In PA-46-350Ps the GAE 43 Altitude Encoder is installed to function as the cabin pressure transducer. It generates cabin altitude and rate information that's collected by the GIA 63Ws and normally displayed on the MFD. It's mounted to the underside of the equipment shelf located forward of the MFD

(b) Removal

1) Disconnect the two avionic harness connectors.

2) Cut safety wire and discard.

3) Unscrew the thumbscrew and slide the unit down and aft to free it from its mounting tray.

(c) Installation

1) Position the unit below and aft of its mounting tray.

2) Engage the lips of the mounting tray and slide the unit forward and up until the thumbscrew can be engaged.

3) Tight thumbscrew to secure unit.

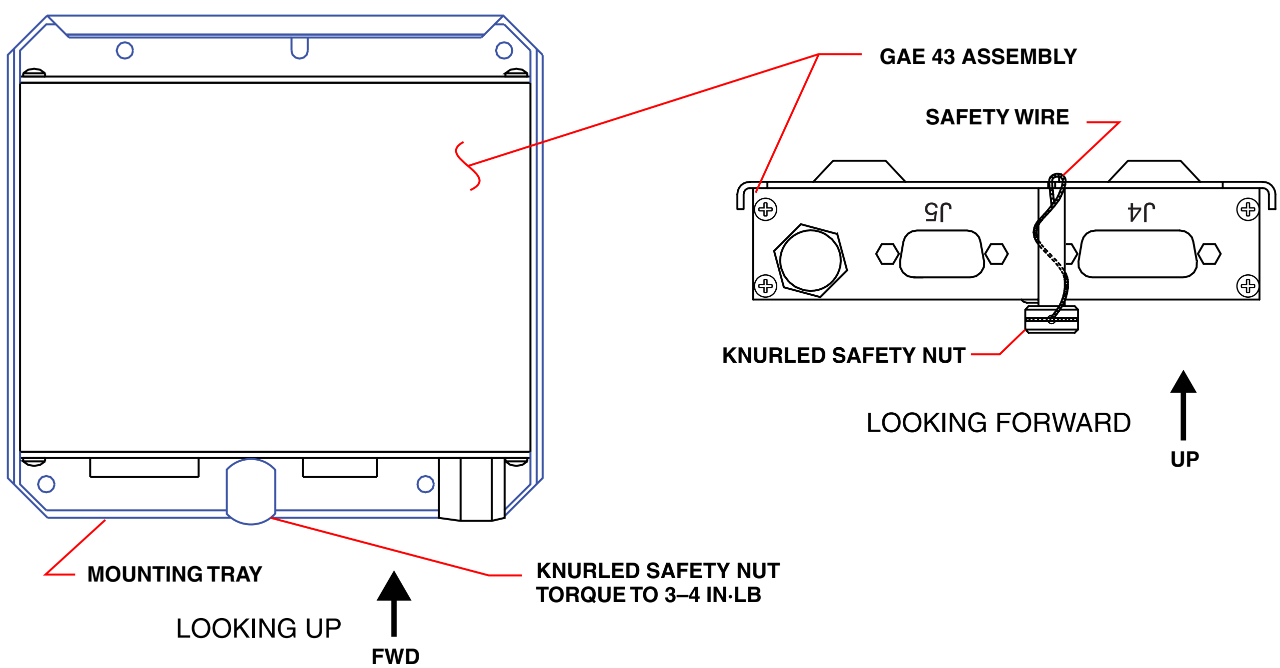
4) Torque per "Figure 19" and safety.

5) Connect the two avionic harness connectors.

(d) Calibration

See "GAE 43 Calibration" on page 342501202 under LRU Test Procedures.

NOTE: Calibrate a new unit before installation.



Effectivity
PA-46-350P

GAE 43 Altitude Encoder
Figure 19

(8) GEA 71 Engine/Airframe Unit

See "Figure 20".

GEA 71 Engine/Airframe Unit
Figure 20

(a) Description

The Garmin GEA 71 Engine/Airframe Unit provides engine/airframe data to the G1000 system. Data received from transducers/sensors is processed and sent to GIA 63s (via RS-485 digital interface), and subsequently to the GDU 1500 MFD. Engine parameters are normally displayed on the MFD. In the event of a MFD failure, all engine instruments can be displayed on the PFDs. The GEA is located behind the instrument panel and is mounted in a vertical orientation. Power is received from the main bus or the Emergency bus.

The GEA interfaces to the following:

- Oil Pressure Sensor
- Oil Temperature Sensor
- Fuel Flow Sensor
- Fuel Filter Sensor
- Fuel Quantity Sendors (left and right).
- Propeller Speed Sensor
- Pitot Heat System
- Flap Position potentiometer.
- Vacuum Transducer.
- Door Ajar switch

(b) Troubleshooting

See "Chart 20" on page 34250198.

(c) Removal

1) Remove PFD2 per "Removal" on page 34250174.
2) Unlock the GEA 71 handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GEA 71.
4) Gently remove the GEA 71 from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GEA 71 into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GEA 71 body using the Phillips screw.
5) Reinstall PFD2 per "Installation" on page 34250174.

CHART 20
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GEA

Failure Message	Cause	Solution
MANIFEST – GEA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GEA 71.	Load the correct software version. See G1000 Software/ Configuration Procedure, below.
GEA1 CONFIG – GEA1 configuration error. Config service req'd.	The system has detected a configuration mismatch in the GEA 71.	Load GEA configuration files. See G1000 Software/ Configuration Procedure, below. If problem persists, replace master configuration module(1), check config module harness for faults and replace if necessary.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

(e) Return to Service

1) Original GEA 71 Reinstalled

No software or configuration loading is required if the removed GEA 71 is reinstalled. Continue to the return-to-service checks.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GEA 71 Installed

If a new, repaired, or exchanged GEA 71 is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 Software/Configuration Procedure" on page 342501150, and then continue to the "GEA 71 Test" on page 342501187.

(9) GEA 71 Backshell Thermocouple

See "Figure 21".

(a) Removal

1) Remove GEA 71 per "Removal" on page 34250197.

2) Remove GEA connector backplate.

3) Remove connector J701 (5) from the backplate.

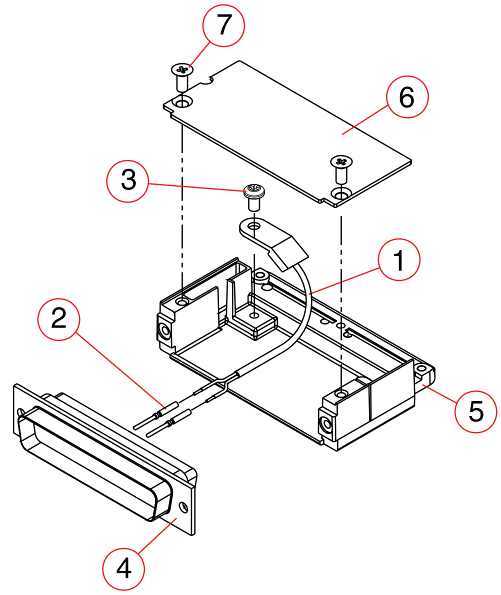
4) Remove cover (6) from the backshell.

5) Unscrew thermocouple from boss on backshell. Extract the thermocouple pins from the connector.

(b) Installation

1) Crimp pins (2) onto each of the thermocouple wires (1). Ensure that pre-stripped wire length is 1/8” prior to crimping.

2) Insert newly crimped pins and wires into the appropriate connector housing location (4) as specified in the electrical schematic.



GEA 71 Backshell Thermocouple
Figure 21

3) Place thermocouple body (1) onto the backshell boss (5). Place the thermocouple as shown in "Figure 21" so that the wires exit towards the bottom of the backshell.

4) Fasten thermocouple tightly to backshell using the provided screw (3).

5) Fasten cover (6) to backshell using the provided screws (7).

6) Reinstall GEA 71 per "Installation" on page 34250197.

(10) GTX 33/33D (ES) Mode S Transponder

See "Figure 22".

(a) Description

The GTX 33/33D provides Mode A, C, and S altitude and position reporting information from the G1000 system. Optional Extended Squitter (ES) units provide additional aircraft flight ID, position, altitude, velocity, climb/descent, and heading information. The units are mounted directly behind the MFD on the center avionics shelf. Power is received from the No.1 and No. 2 avionics buses respectively. Similarly to the GEA 71, the GTX 33 units send data via RS-232 directly to the on-side GIA 63W. Information is then sent to the PFD, where the pilot can control the transponder. The GTX 33 units interface with the transponder antenna.

GTX 33 Mode S Transponder
Figure 22

(b) Troubleshooting

See "Chart 21" on page 342501101.

(c) Removal

1) Remove the MFD per "Removal" on page 34250174.
2) Unlock the GTX 33 handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GTX 33. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert GTX 33 into its rack. The handle should engage dogleg track.
3) Press down on the GTX 33 hand to lock the unit into the rack.
4) Lock the handle to the GTX 33 body using the Phillips screw.
5) Reinstall the MFD per "Installation" on page 34250174.

(e) Return to Service

1) Original GTX 33 is Reinstalled

No software or configuration loading is required if the removed GTX 33/33D is reinstalled. Continue to "GTX 33/33D (ES) Testing" on page 342501210

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GTX 33 Installed in Opposite Locations for Troubleshooting

No software loading is required if the original GTX1 and GTX2 are installed in opposite locations. Continue to "Aircraft Registration Number Entry (Transponder Configuration)" on page 342501157. And "GTX 33/33D (ES) Testing" on page 342501210.

3) New, Repaired, or Exchanged GTX 33 is Installed

If a new. repaired. or exchanged GTX 33 is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 Software/Configuration Procedure" on page 342501150, then continue to "Aircraft Registration Number Entry (Transponder Configuration)" on page 342501157. And "GTX 33/33D (ES) Testing" on page 342501210.

CHART 21
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GTX 33/33D

Failure Message	Cause	Solutions
MANIFEST – GTX1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GTX 33.	• Reload software. See Software Load / Configuration Procedure.
MANIFEST – GTX2 software mismatch. Communication halted.		
XPDR1 CONFIG – XPDR1 configuration error. Config service req'd.	The system has detected a configuration mismatch for the specified GTX 33.	• Perform a SET>ACTV configuration reset on the GTX Config page and verify the aircraft registration is present. • If error is still present, reload config files from a loader card. ✓ If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.
XPDR2 CONFIG – XPDR2 configuration error. Config service req'd.		
XPDR1 SRVC – XPDR1 needs service. Return unit for repair.	The G1000 has detected a failure in the specified GTX 33.	• Replace GTX 33/33D.
XPDR2 SRVC – XPDR2 needs service. Return unit for repair.		
XPDR1 FAIL – XPDR 1 is inoperative.	The specified GTX 33 is not responding.	• Check wiring between GIA's and GTX. • Replace GTX 33/33D.
XPDR2 FAIL – XPDR 2 is inoperative.		

(11) GDC 74A Digital Air Computer

See "Figure 23".

(a) Description

The GDC 74A computers compile information from the pitot/static system and the GTP 59 outside air temperature (OAT) probe to digital air data computations to the G1000 system. The GDC 74A communicates with the GIA 63W, GDU 1040, and GRS 77 using ARINC 429 digital interface. The units are mounted behind the MFD. Power is provided to #1 GDC 74A unit from the main bus or the emergency bus. The #2 GDC 74A is powered by the #2 avionics bus.

GDC 74A Digital Air Computer
Figure 23

(b) Troubleshooting

See "Chart 22" on page 342501103.

(c) Removal

1) Remove the MFD per "Removal" on page 34250174.
2) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.
3) Loosen each thumbscrew on the hold-down clamp and remove the clamp.
4) Carefully remove the unit from its mount.

(d) Installation

1) Place the unit in the mounting tray.
2) Position the locking clamp and fasten using the thumbscrews.
3) Connect the pitot/static plumbing.
4) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.
5) Reinstall the MFD per "Installation" on page 34250174.

(e) Return to Service

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 341014

1) Original GDC 74A is Reinstalled

No software or configuration loading is required if the removed GDC 74A is reinstalled. Continue to "GDC 74A Air Data Computer Tests" on page 342501188.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GDC 74A Installed in Opposite Locations for Troubleshooting

No software loading is required if the original GDC1 and GDC2 are installed in opposite locations. Continue to "GDC 74A Air Data Computer Tests" on page 342501188.

3) New, Repaired, or Exchanged GDC 74A is Installed

If a new, repaired, or exchanged GDC 74A is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 Software/Configuration Procedure" on page 342501150, and then continue to "GDC 74A Air Data Computer Tests" on page 342501188.

4) New GDC 74A Configuration Module is Installed

The correct configuration files must be loaded if the GDC 74A configuration module has been replaced.

See "G1000 Software/Configuration Procedure" on page 342501150, and then continue to "GDC 74A Air Data Computer Tests" on page 342501188.

CHART 22
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GDC 74A

Symptom	Recommended Action
Altitude is different than standby altimeter.	Perform a pitot/static check per Pitot Static System Test in 34-10-00. Allow the GDC to warm up for fifteen minutes before checking accuracy. Determine which instrument is outside limits and recalibrate or replace. Note: Both units may individually be in spec but show a difference in altitude. Do not return the GDC to Garmin for service if not outside limits.
GDC Config file does not load.	Replace GDC config module. If problem persists, replace GDC config module wire harness.

Failure Message	Cause	Solution
MANIFEST – GDC1 software mismatch Communication halted. MANIFEST – GDC2 software mismatch Communication halted.	The system has detected an incorrect software version loaded in the specified GDC.	Load correct software version. See G1000 Software/ Configuration Procedure, below.

(12) GTP 59 OAT Probe

See "Figure 24".

(a) Description

The Garmin GTP 59 OAT Probe provides the GDC 74A with air temperature data. The OAT probes are mounted to an access cover on the left wing.

(b) Troubleshooting

See "Chart 5" on page 34250131.

(c) Removal

1) Remove the access cover.
2) Disconnect the connector.
3) Use an open-end wrench to hold the probe in place on the inside of the access cover. On the outside of the access cover, loosen the GTP 59 Housing and remove the GTP 59.

GTP 59 OAT Probe
Figure 24

(d) Installation

1) Place the GTP 59 grounding strap O-ring connector over the OAT probe end and insert the probe through the access cover from the inside side. Secure with housing and washer on the outside of the access cover. Use an open-end wrench to hold the GTP 59 on the inside and torque the housing on the outside to 50 ± 5 in. lbs.
2) Reconnect the connector to the airplanes wiring harness.
3) Install the access cover.

(13) GRS 77 Attitude and Heading Reference System

See "Figure 25".

(a) Description

The Garmin GRS 77 AHRS units provide attitude and heading information to the G1000 system. The units, mounted behind the pilots seat, contain advanced tilt sensors, accelerometers, and rate sensors. The unit interfaces with the GDC 74A and GMU44 Magnetometer and utilizes GPS signals from the GIA 63Ws. Actual attitude and heading information is sent using ARINC 429 digital interface to both GDU 1040s and GIA 63Ws. Power is provided to #1 GRS 77 unit from the main bus or the emergency bus. The #2 GDC 74A is powered by the #2 avionics bus. The GRS 77 interfaces with and provides power to the GMU 44 Magnetometer. The GRS 77 supplies attitude and heading information directly to the PFDs, MFD, and GIAs.

GRS 77 AHRS
Figure 25

NOTE: GRS 77 Earth Magnetic Field Updates

The GRS 77 utilizes an Earth magnetic field model which is updated once every five (5) years. The update should be available from Garmin in each of the following years: 2020, 2025, and every five (5) years thereafter, as long as the GRS 77 remains a Garmin-supported product. The G1000 system alerts the operator that the magnetic field database is out of date by issuing the message “AHRS SERVICE – AHRS Magnetic-field model needs update”. Garmin will distribute update instructions when updates are available.

(b) Troubleshooting

See also "Chart 23" on page 342501108 and "Chart 24" on page 342501109.

1) Review the airframe logbook. See if any G1000 or other avionics or electrical maintenance had been performed recently that might produce or have induced the trouble being observed.
2) Check power wire connections at the circuit breakers. Inspect for loose wire terminals causing intermittent power glitches. Also, check for intermittent circuit breakers.
3) Have ground power put on the aircraft.
4) Turn on G1000 and record the system software level on the MFD. This will be needed if assistance from Garmin Product Support is required.
5) After the G1000 system has initialized (about one minute from power on), note any Red-X’s on the displays, ALERT messages and Red-X’s on the MFD Aux – System Status page.
6) Try to verify the issue still exists before proceeding to the physical inspection below.
7) Turn off G1000 and gain access to the GRS77.
8) Inspect the physical installation of the GRS.
   a) Is the connector tight and locking slider engaged to the locking tabs on each side of the GRS connector?
   b) Is the wire harness loose and able to move around during flight? This condition may cause the wire to pull on or vibrate the connector making intermittent connections.
   c) Is the GRS mounted tight to the rack? If any doubt exists, use a screwdriver to check the tightness of the four mounting screws.
   d) Look around the GRS for any heavy objects that may not be fastened tight to the structure that could induce GRS vibration.

e) Look for evidence of water or fluid contamination in the area around the GRS.

f) Unplug the GRS connector and check for bent pins.

g) Inspect the wire harness clamp on the rear of the connector to verify it's not too tight and damaging wires. Also check for some sort of protective wire wrap between the wires and the clamp. If the wire clamp is installed upside down, it has sharp edges that can cut into the wires.

h) Verify the locking slider spring is strong enough to keep the slider in the locked position by cycling the slider.

i) Ensure that no cell phone or other device using cell phone technology is turned on (even in a monitoring state) in the cabin.

j) Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements.

k) Check PFD1 Alert Window for PFD1/2, MFD or GRS configuration, software or failed data path error messages. Correct any errors before proceeding.

l) Ensure GRS 77 unit connecter is secure and proper wire harness strain relief is provided.

m) Ensure the GRS 77 is fastened down tightly in its mounting rack and that the mounting rack is not loose.

NOTE: Do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be releveled and and GRS 77 recalibrated.

9) Cycle GRS 77 power to restart initialization.

10) Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important for an ATTITUDE FAIL that appears during ground operation only.

11) Perform "Engine Run-Up AHRS Vibration Test (Procedure C)" on page 342501195 to check if engine vibration is causing the GRS 77 to invalidate outputs or reset.

12) Perform "Magnetometer Interference Test" on page 342501196 to determine if there is significant magnetic interference from some device on the aircraft.

13) Replace GRS 77.

14) If problem persists, replace the GRS77 Configuration module.

15) If the condition is not resolved by following these instructions, contact Garmin Product Support for additional assistance. In rare cases, a Garmin Field Service Engineer may need to visit your facility to retrieve the fault logs stored in the GRS to determine if the fault is in the GRS or the aircraft.

(c) Removal

1) Disconnect the AHRS connector.

2) Remove the four Phillips thumbscrews with a screwdriver and set them aside.

3) Gently lift the GRS 77 from the mounting plate.

NOTE: If the mounting plate itself is removed or loosened, the GRS 77 must be recalibrated upon reinstallation.

(d) Installation

1) Place the GRS 77 on the mounting plate, ensuring the orientation is correct.

2) Fasten the unit to the plate using the Phillips thumbscrews.

3) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 77.

(e) Return to Service

1) Original GRS 77 is Reinstalled

If the original GRS 77 was reinstalled, then no software loading is required.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

If the GRS rack was not removed or loosened, continue to "GRS/GMU Test" on page 342501198.

If the GRS rack was removed or loosened, continue to "GRS 77 / GMU 44 Calibration Procedures" on page 342501190.

2) Original GRS 77 Installed in Opposite Locations for Troubleshooting

If the original GRS1 and GRS2 are installed in opposite locations (GRS1 and GRS2 in opposite unit racks), no software loading is required.

If the GRS rack(s) was not removed or loosened, continue to "GRS/GMU Test" on page 342501198.

If the GRS rack(s) was removed or loosened, continue to "GRS 77 / GMU 44 Calibration Procedures" on page 342501190.

3) New, Repaired, or Exchanged GRS 77 is Installed

If a new, repaired, or exchanged GRS 77 unit is installed then software must be loaded and the unit must be calibrated. Continue to "G1000 Software/Configuration Procedure" on page 342501150 for software loading. Then, proceed as follows:

If the GRS rack(s) was not removed or loosened, continue to "GRS/GMU Test" on page 342501198.

If the GRS rack(s) was removed or loosened, continue to "GRS 77 / GMU 44 Calibration Procedures" on page 342501190.

4) New GRS 77 Configuration Module is Installed

If the GRS 77 configuration module was replaced, no software loading is required. Continue to "GRS 77 / GMU 44 Calibration Procedures" on page 342501190.

CHART 23
G1000 - TROUBLESHOOTING - GRS 77 (AHRS) / GMU 44

Symptom	Recommended Action
AHRS does not complete initialization	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important if this issue appears during ground operation only. • Calibrate the GRS 77. • Check GRS 77 configuration module wiring for damage. • Check GRS 77 connector for bent pins. ✓ If no damage can be found, replace GRS 77 configuration module. ✓ If problem persists, replace the GRS 77.
Attitude appears unstable	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure the four GRS 77 mounting screws are tight. Finger tight is not sufficient, a screwdriver must be used to verify. • Ensure mounting rack and airframe shelf are secure and all hardware and brackets are present (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Ensure GRS 77 connector is securely fastened and proper strain relief is provided. • Remove GRS 77 connector and verify there are no bent pins. • Replace the GRS 77. • Contact Garmin for further troubleshooting if required.

CHART 24 (Sheet 1 of 2)
G1000 - MESSAGE ADVISORIES - GRS / GMU

Failure Message	Cause	Solution
MANIFEST – GRS1 software mismatch. Communication halted. MANIFEST – GRS2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GRS 77.	Load correct software version. See G1000 Software/ Configuration Procedures, below.
AHRS1 SRVC – AHRS1 magnetic-field model needs update. AHRS2 SRVC – AHRS2 magnetic-field model needs update.	The AHRS magnetic field model in the affected unit is out-of-date. Appears on ground only.	Load updated AHRS magnetic field file. See Garmin SB 533, latest revision.
GEO LIMITS – AHRS1 too far north/south, no magnetic compass. GEO LIMITS – AHRS2 too far north/south, no magnetic compass.	No magnetic compass information available due to airplane being too far north or south.	Operate the aircraft only within the geographic limits specified in the POH.
AHRS1 TAS – AHRS1 not receiving airspeed. AHRS2 TAS – AHRS2 not receiving airspeed.	The specified GRS 77 is not receiving airspeed from the GDC 74A.	Check GRS/GDC interconnect for faults. Replace the GDC 74: If problem persists, replace the GRS 77.
AHRS1 GPS – AHRS1 not receiving backup GPS information.	The GRS 77 #1 is not receiving backup GPS information from either GIA 63.	Ensure that a cell phone or a device using cell phone technology is not turned on
AHRS1 GPS – AHRS1 operating exclusively in no-GPS mode.	The GRS 77 #1 is operating in the absence of GPS.	(even in a monitoring state) in the cabin.
AHRS1 GPS – AHRS1 not receiving any GPS information.	The GRS 77 #1 is not receiving GPS data from the GPS receivers.	Check GPS status for GIA 1 and 2 on MFD - AUX GPS
AHRS1 GPS – AHRS1 using backup GPS source.	The GRS 77 #1 is using the backup GPS data path.	STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting chart. Troubleshoot GIA1/2 –GRS1 wiring for faults per “Failed Path Messages.” Replace the GRS 77 #1.

CHART 24 (Sheet 2 of 2)
G1000 - MESSAGE ADVISORIES - GRS / GMU

Failure Message	Cause	Solution
AHRS2 GPS – AHRS2 not receiving backup GPS information.	The GRS 77 #2 is not receiving backup GPS information from either GIA 63.	Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. Check GPS status for GIA 1 and 2 on MFD - AUX GPS STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting chart. Troubleshoot GIA1/2 –GRS2 wiring for faults per “Failed Path Messages.” Replace the GRS 77 #2.
AHRS2 GPS – AHRS2 operating exclusively in no-GPS mode.	The GRS 77 #2 is operating in the absence of GPS.	
AHRS2 GPS – AHRS2 not receiving any GPS information.	The GRS 77 #2 is not receiving GPS data from the GPS receivers.	
AHRS2 GPS – AHRS2 using backup GPS source.	The GRS 77 #2 is using the backup GPS data path.	
AHRS MAG DB – AHRS magnetic model database version mismatch.	The G1000 has detected a magnetic model database version mismatch.	Load updated AHRS magnetic field file in both units. See Garmin SB 533, latest revision.
MANIFEST – GMU1 software mismatch. Communication halted. MANIFEST – GMU2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GMU 44.	Load the correct software version. See G1000 Software/Configuration Procedure, below.
HDG FAULT – AHRS1 magnetometer fault has occurred. HDG FAULT – AHRS2 magnetometer fault has occurred.	A fault has occurred in the specified magnetometer; heading will be flagged invalid.	Check GMU 44/GRS 77 interconnect for faults. Replace GMU 44. If problem persists, replace affected GRS 77.

(14) GMU 44 Magnetometer

See "Figure 26" and "Figure 27" on page 342501112.

(a) Description

The GMU 44 provides horizontal and vertical magnetic field information to the GRS 77 AHRS. This allows heading to be calculated and provides assistance during AHRS alignment. The GMU 44 is mounted under the dorsal fairing atop the aft fuselage at F.S. 282.372.

GMU 44 Magnetometer
Figure 26

(b) Troubleshooting

See "Chart 23" on page 342501108 and "Chart 24" on page 342501109.

(c) Removal

1) Remove the dorsal fairing to expose the magnetometers.
2) Disconnect the wiring harness(es) from the aft fuselage harness.
3) Unscrew and remove the three brass screws and washers securing the magnetometer to its mounting bracket and remove the magnetometer(s).
4) Carefully lift the GMU 44 from the rack.

(d) Installation

CAUTION: THE MAGNETOMETERS ARE SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Lower the GMU 44 into the rack and secure the plate with the three Phillips screws.
3) Connect wiring harness(es) to the aft fuselage harness.
4) Reinstall the dorsal fairing.

NOTE: Ensure correct hardware is used when reinstalling vertical fin dorsal fairing over magnetometers.

(e) Return to Service

1) Original GMU 44 is Reinstalled

If the original GMU 44 was reinstalled, then no software loading is required. Continue to "GRS/GMU Test" on page 342501198.

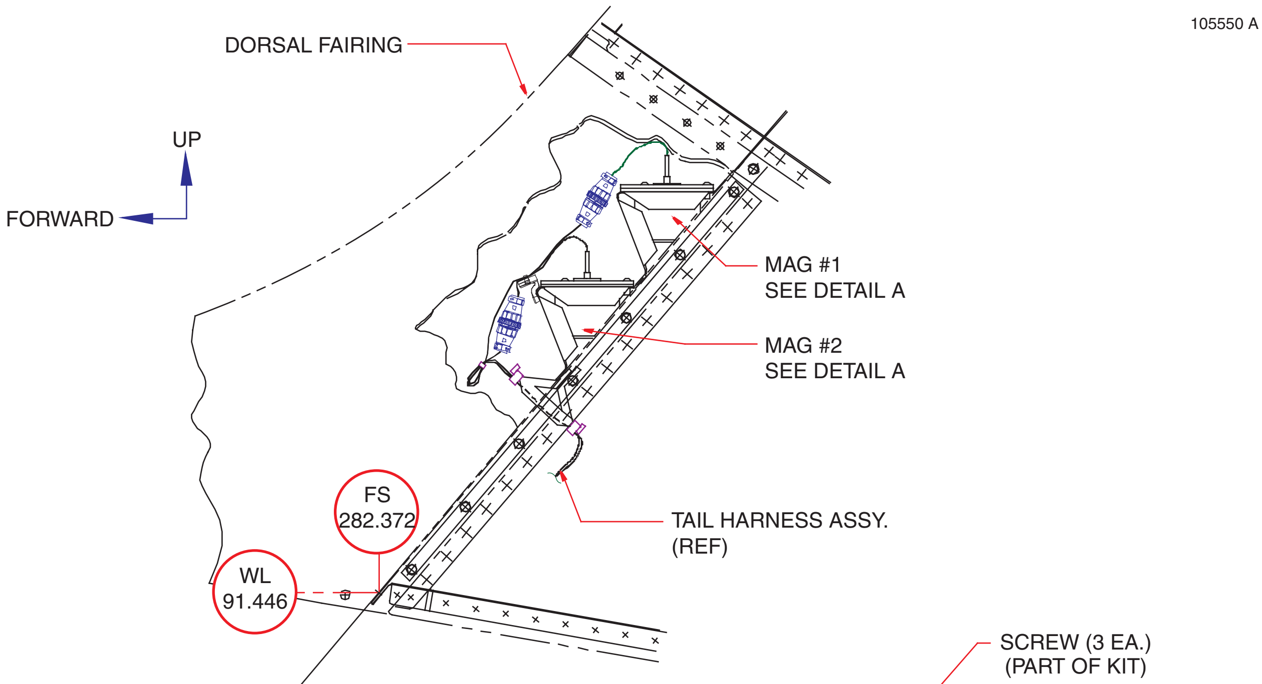
NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

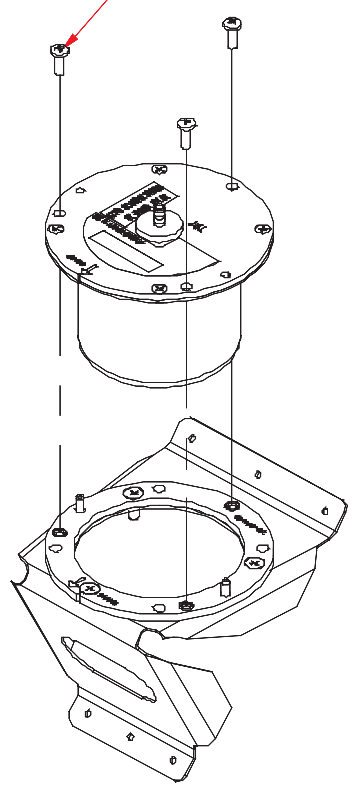
2) Original GMU 44 Installed in Opposite Locations for Troubleshooting

If the original GMU1 and GMU2 are installed in opposite locations (GMU1 and GMU2 in opposite unit racks), no software loading is required. However, performing "GRS 77 / GMU 44 Calibration Procedures" on page 342501190 is required. Then, continue to "GRS/GMU Test" on page 342501198.

3) New, Repaired, or Exchanged GMU 44 is Installed

If a new, repaired, or exchanged GMU 44 unit is installed then software must be loaded. Continue to "G1000 Software/Configuration Procedure" on page 342501150. Then continue to "GRS 77 / GMU 44 Calibration Procedures" on page 342501190.





CAUTION: ENSURE ONLY BRASS SCREWS ARE USED TO SECURE MAGNETOMETERS TO MOUNTING BRACKETS.

NOTE: ENSURE CORRECT HARDWARE IS USED WHEN INSTALLING DORSAL FAIRING.

DETAIL A
MAGNETOMETER INSTALLATION
(2 PLCS)

Magnetometer/OAT Sensor Assemblies
Figure 27

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(15) GDL 69A/69eA Satellite Datalink Unit (Optional)

See "Figure 28".

NOTE: The GDL 69A was factory installed 4636633, 4636652–4636715. The GDL 69eA was factory installed in 4636717–4636719 and is an authorized service replacement in earlier airplanes.

(a) Description

The GDL 69A/69eA provides XM Radio weather and music entertainment through means of a dedicated satellite datalink. The GDL 69A/69eA is mounted behind the PFD1. Power to the GDL 69A/69eA is received from the #1 avionics bus. The GDL 69A/69eA sends weather data through the HSDB bus to the MFD, where the datalink interface is controlled. Digital audio is sent directly to the GMA 347 or GMA 350 Audio Panel.

GDL 69A Datalink Unit (Shown)
Figure 28

(b) Troubleshooting

See "Chart 25" on page 342501115 and "Chart 26" on page 342501116.

(c) Removal

1) Remove PFD1 per "Removal" on page 34250174.
2) Unlock the GDL 69A/69eA handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GDL 69A/69eA.
4) Gently remove the GDL 69A/69eA from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GDL 69A/69eA into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GDL 69A/69eA body using the Phillips screw.
5) Reinstall PFD1 per "Installation" on page 34250174.

(e) Return to Service

1) Original GDL 69A/69eA is Reinstalled

No software or configuration loading is required if the removed GDL 69A/69eA is reinstalled. Continue to "GDL 69eA Test" on page 342501199.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GDL 69A/69eA is Installed

If a new, repaired, or exchanged GDL 69A/69eA is installed, the correct software and configuration files must be loaded to the unit, then the XM Satellite Radio subscription must be reactivated, see "Chart 25" on page 342501115. Then continue to "GDL 69eA Test" on page 342501199.

CHART 25
G1000 - TROUBLESHOOTING - GDL 69A/69eA

Symptom	Recommended Action
No XM audio is heard	Ensure the following items are not preventing the audio panel from distributing XM audio (see also applicable Garmin G1000 Pilot's Guide).
No XM weather information is displayed.	Verify XM volume is not muted on the AUX–XM RADIO page on the MFD.
	Ensure nothing is plugged into the Stereo Input Jack.
	Verify the COM squelch is not open.
	Verify the ICS squelch is not open.
	Verify the marker beacon tones are not being received.
	Verify the headphone (if equipped) volume is turned up.
	Go to the AUX – SYSTEM STATUS page on the MFD and ensure unit is online.
	If a red X is present, verify the unit is receiving power at the rack connector.
	Ensure there are no GDL alerts in the alert window. If there is an Alert for software or configuration error or mismatch, reload the file noted in the Alert.
	Restart the PFDs and MFD in configuration mode and go to the GDL page.
	Verify unit is active.
	Verify the Signal number is "2" or "3". If it is "0" or "1" check the GDL antenna and cabling for faults.
	Reseat the GDL to verify the coax connector is fully seated.
	If unit is not active, contact XM Customer service at 1-800-985-9200 to have a refresh signal sent to your unit.
	NOTE: You will need to provide them the Audio Radio ID (and Data Radio ID number for XM weather) numbers. Also verify with XM that the correct Weather package (Aviator Lite or Aviator) is on the account, and that no traffic service has been activated against that Radio ID. The unit must be on for approximately one hour after the request for the refresh has been sent to receive the signal. To obtain radio IDs, power-up PFD1 in normal mode and then select XM INFO page in AUX page group. XM weather services has an 8 digit "data radio ID" and XM radio services has an 8 digit "audio radio ID" displayed on this page.
	Alternatively, you may also go to XM's website at http://www.siriusxm.com/refresh and enter the radio ID's to have a refresh signal sent.
	If there is still problems receiving weather products after performing the above step, call XM and have the account deactivated, and a new account activated to clear out any corrupt account information.
	Verify there is a good ground connection through the aircraft between the MFD and the GDL unit. See 51-80-00 for bonding checks.
	If problem persists, replace the GDL.

CHART 26
G1000 - MESSAGE ADVISORIES - GDL 69A/69eA

Failure Message	Cause	Solution
GDL 69 FAIL – GDL 69A/69eA has failed.	The G1000 has detected a failure in the GDL 69A/69eA.	Replace GDL 69A/69eA. Check GDL 69A antenna and cabling. Check the GDL 69A/69eA and MFD interconnect.
GDL69 CONFIG – GDL 69A/69eA configuration error. Config service req'd.	The G1000 has detected a configuration mismatch in the GDL 69A/69eA.	Reload configuration files(1). If problem persists, replace master configuration module(2), check config module harness for faults and replace if necessary.
MANIFEST – GDL software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GDL 69A/69eA.	Load correct software version (1).

(2) "G1000 Software/Configuration Procedure" on page 342501150.
(3) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e., TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

(16) GRT 10 Wireless Transceiver Unit (Optional)

See "Figure 29".

(a) Description

The GRT 10/GRC 10 wireless remote system is for use by passengers in the aircraft to control the audio functions (audio volume and channel changes) of the Garmin GDL 69A/69eA Datalink Receiver. The system consists of two components:

1) The GRT 10 Wireless Transceiver installed in the aircraft and connected to the GDL 69A/69eA serial port, and

2) the GRC 10 Wireless Remote with an LCD display.

(b) Removal

1) Disconnect backshell assembly from unit.

2) Use a 3/16” hex drive tool to remove each of the four mounting screws.

(c) Installation

1) Inspect connector for damaged pins.

2) Hold unit flush with the mounting plate.

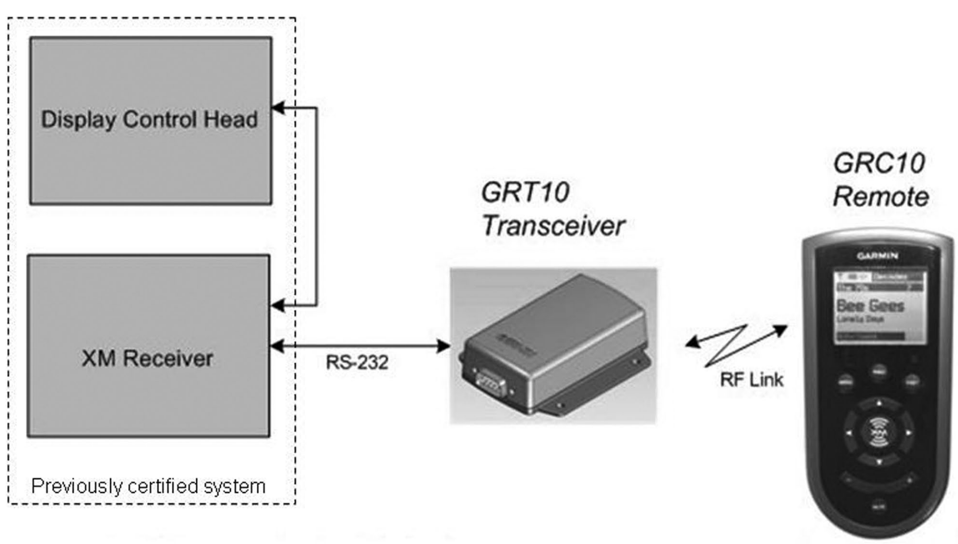
3) Use a 3/16” hex drive tool to tighten each of the four mounting screws.

4) Connect backshell assembly to unit.

(d) Return to Service

If either the GRT 10 or the GRC 10 is replaced:

Perform "GRC 10 XM Pairing" on page 342501156.



GRT 10
Figure 29

(17) GSR 56 Iridium Satellite Transceiver

Optional in S/N's 4636633, 4636652 and up.

See "Figure 30".

(a) Description

The GSR 56 Iridium Satellite Transceiver provides airborne Iridium Satellite Telephone and SMS Messaging service. Iridium telephone and text meassaging are available to the flight crew through the MFD, audio panel, and headset. The Garmin GSR 56 is connected to the #2 GIA 63W and GMA 350. Power is received from the Avionics Bus #1.

(b) Troubleshooting

See "Chart 27" on page 342501119.

(c) Removal

1) Remove the cabin rear closeout panel to gain access to the upper aft avionics shelf.

2) Loosen the GSR 56 unit by turning the ratchet mechanism located at the front of the unit counter-clockwise until it drops free of the locking pawl on the unit.

3) Gently remove the GSR 56 from its rack.

GSR 56 Iridium Transceiver
Figure 30

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

2) Gently insert the GSR 56 into the rack and engage the connectors.

3) Lift the ratchet mechanism to allow the collar to engage the locking pawl on the unit.

4) Gently turn ratchet mechanism clockwise to secure the GSR 56 unit into place.

5) Reinstall the cabin rear closeout panel.

(e) Return to Service

The GSR 56 does not require software or configuration loading if replaced. Instead, Garmin and/or Iridium need to be contacted to establish service. See "Chart 27" on page 342501119.

CHART 27
TROUBLESHOOTING / MESSAGE ADVISORIES - GSR 56

Problem	Action
No communication with GSR 56.	Check power wiring and pin out. Verify correct communication port setting on display/control device.
No or low-quality signal.	Ensure the Iridium antenna has an unobstructed view of satellite constellation. Check the antenna cable and connectors. Verify antenna ground plane is adequate.
No audio output.	Check wiring from GSR 56 to audio panel or GDL. Verify subscription with Garmin Iridium Services, http://fly.garmin.com or 1-866-739-5687.
Unable to make a phone call.	Verify subscription with Garmin Iridium Services, see above. Verify signal quality is adequate. Verify communication between GSR 56 and the display/control device.

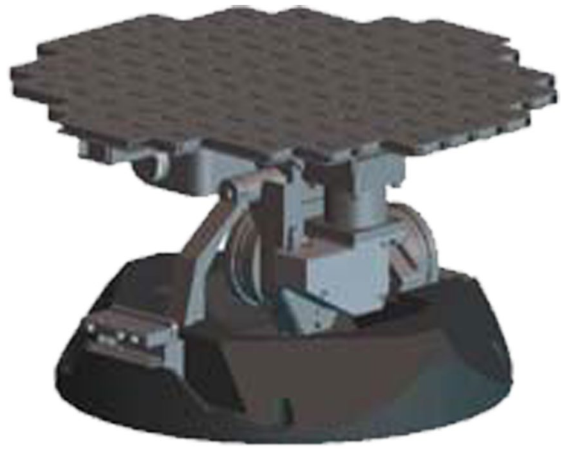
Failure Message	Cause	Solution
GSR1 FAIL – GSR1 is inoperative.	The G1000 has detected a failure in the GSR 56.	Check wiring for faults per “Failed Path Messages.” Replace the GSR 56.

(18) GWX 68 Weather Radar

See "Figure 31".

(a) Description

The GWX 68 Airborne Weather Radar provides weather radar data output to the MFD. The GWX 68 is mounted in the right wing radar pod. Power to the GWX 68 is received from the avionics No. 1 bus. Data received from the GWX 68 is routed through the GDL data link unit to the MFD via high-speed data bus (Ethernet). If the optional GDL is not installed, the GWX Ethernet is connected directly to the MFD.



GWX 68 Weather Radar
Figure 31

(b) Troubleshooting

See "Chart 28" on page 342501121.

(c) Removal

1) Remove screws and washers (13 ea.) and remove radome.
2) Disconnect backshell assembly from unit.
3) Use a 3/16” hex drive tool to remove each of the four mounting screws.

(d) Installation

1) Inspect connector for damaged pins.
2) Hold unit flush with the radar mount.
3) Use a 3/16” hex drive tool to tighten each of the four mounting screws. Torque for 1/4-28 fasteners per Gap Conditions Between Parts Attached with Threaded Fasteners under Torque Requirements in 91-10-00, except final torque not to exceed 60 in.-lbs.
4) Connect backshell assembly to unit.
5) Install radome and secure with screws and washers (13 ea.).

(e) Return to Service

1) Original GWX 68 Reinstalled

No software or configuration loading is required if the removed GWX 68 is reinstalled, continue to "GWX 68 Test" on page 342501201.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GWX 68 Installed

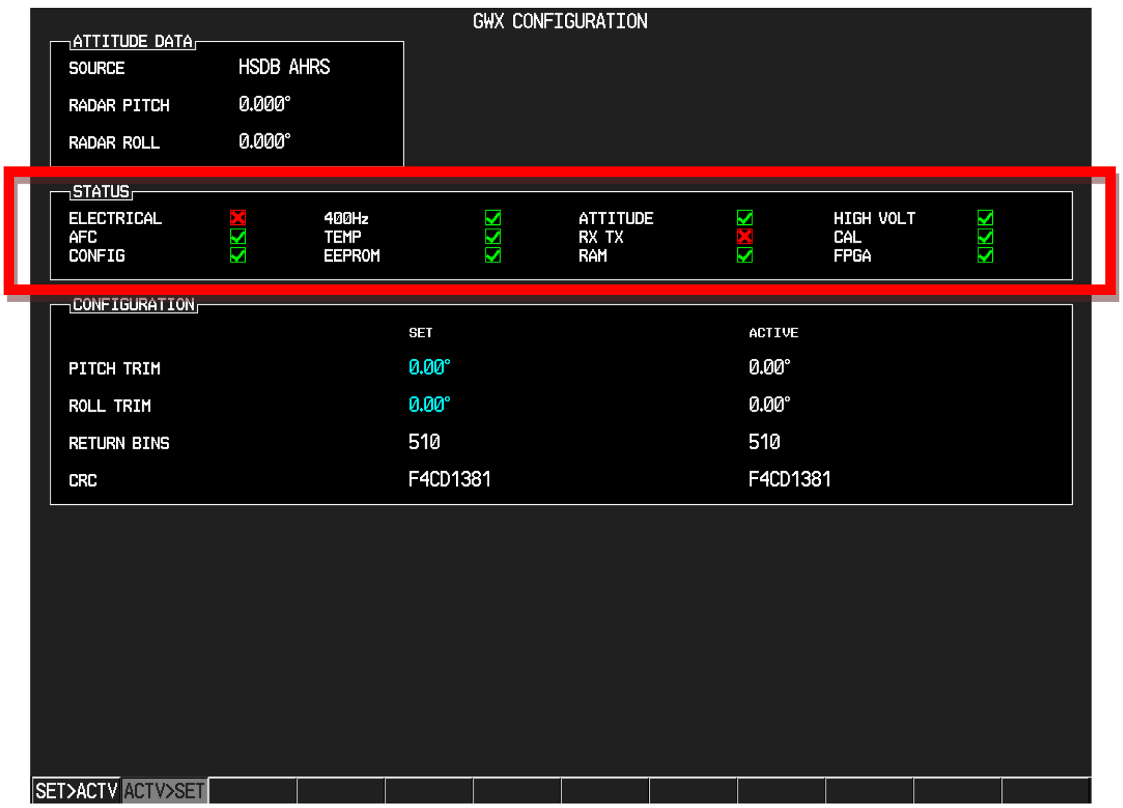
If a new, repaired, or exchanged GWX 68 is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 Software/Configuration Procedure" on page 342501150 and then continue to "GWX 68 Test" on page 342501201.

CHART 28 (Sheet 1 of 5)
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68

WARNING: BEFORE ENERGIZING THE EQUIPMENT, BE SURE MICROWAVE RADIATION SAFETY PRECAUTIONS INCLUDING BOTH FUEL AND PERSONNEL SAFETY CONSIDERATIONS HAVE BEEN OBSERVED. THESE INCLUDE CLEARING ALL PERSONNEL TO AN AREA BEYOND THE MAXIMUM PERMISSIBLE EXPOSURE LEVEL (MPEL) BOUNDARY. THE MPEL FOR THE GWX 68 IS 11 FEET.

GWX Configuration Page Data Fault Indications



NOTE: The fault indications are reset if PFD1 is transitioned from normal mode to configuration mode. It will return if the fault is persistent. It may not return if the fault is intermittent. The fault indications also reset with a power cycle to the unit.

NOTE: Transitioning the MFD to configuration mode will not reset the fault indications.

CHART 28 (Sheet 2 of 5)
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68

STATUS INDICATOR BOX	FAULT INDICATION		FAULT INDICATION DESCRIPTION	CORRECTIVE ACTION
ELECTRICAL			Voltages within the unit are out of range.	• Cycle unit twice to confirm problem. • Replace the GWX 68 if condition remains.
400Hz				• N/A for G1000 installations, this box should always be green.
HIGH VOLT			Monitoring for >33V at the power input to the GWX. Transitions unit to standby mode if over 33V.	• Check voltage at input of unit (should be <33V). If over 33V, troubleshoot aircraft power system. • Check if condition occurred due to aircraft system power spike due to other non-G1000 electrical equipment loads. • Correct issues that cause aircraft electrical system power spikes. • If problem persists, replace the GWX. • Cycle power on unit only to see if fault remains, if it does, replace the GWX.
AFC			Automatic frequency control within the unit has a fault, will make unit enter Standby mode.	• If problem persists, replace the GWX.
TEMP			Unit temp >85° C.	• If condition occurred during flight, replace the GWX 68. • If condition occurs on the ground due to excessive heat inside the radome, turn unit off and allow it to cool before further use. • If the condition occurs on the ground and the temperature is normal in the radome, replace the GWX 68.
RX TX			Magnetron fault within the unit, will make unit enter Standby mode.	• Ensure GWX software version 2.10 or later is installed. • If problem persists, replace the GWX.

CHART 28 (Sheet 3 of 5)
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68

STATUS INDICATOR BOX	**FAULT INDICATION**		**FAULT INDICATION DESCRIPTION**	**CORRECTIVE ACTION**
CAL			Bad factory calibration data.	Replace the GWX 68.
CONFIG			GWX and GDU configuration do not match.	Troubleshoot per GWX config message below.
EEPROM			Internal EEPROM component failed.	Replace the GWX 68.
RAM			Internal RAM component failed.	Replace the GWX 68.
FPGA			Internal FPGA component failed.	Replace the GWX 68.
ATTITUDE			The unit does not have a valid attitude source via A429 databus.	• Check the AHRS source for proper operation. • If problem persists, replace the GWX.

CHART 28 (Sheet 4 of 5)
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68

Symptom	Possible Cause	Resolution
Radar audibly noisy during ground operation.	Excessive gear train noise or radome interference.	Remove radome. Inspect interior surfaces of the radome for contact with the radar dish. Correct interference by adjusting or replacing radome. If no radome interference is found, perform the following: Turn on the G1000. Select the radar page on the MFD. Press the following softkeys in the following order: 7, 9, 9, 7. Press the MODE softkey Press the TEST softkey. The radar dish will begin scanning, but it will not be transmitting. Test the horizontal and vertical modes to exercise the gear train and evaluate for noise. Replace the GWX 68 if the gears are excessively noisy.
Radar fails to make a full vertical or horizontal sweep on the screen.	Radome Interference or GWX 68 failure.	Remove the radome and inspect interior surfaces for contact with the radar dish. Correct interference by adjusting or replacing the radome. If problem persists, replace the GWX 68.

CHART 28 (Sheet 5 of 5)
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68

Failure Message	Cause	Solution
GWX CONFIG – GWX configuration error. Configuration service required.	The G1000 has detected a GWX 68 configuration mismatch.	Load GWX configuration files. Replace GWX. If problem persists, replace master configuration module(1), check config module wiring for faults and replace if necessary.
GWX FAIL – GWX is inoperative.	The G1000 has detected a failure in the GWX 68.	Check Ethernet connection between the GWX and GDL69A for faults. Replace the GWX 68.
GWX SERVICE – Needs service. Return unit for repair.	The G1000 has detected a failure in GWX 68.	Replace the GWX 68.
MANIFEST – GWX software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GWX 68.	Load correct software version. See G1000 Software/ Configuration Procedure, below.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

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(19) GTS 825 Traffic Advisory System

Optional in S/N's 4636633, 4636652 and up.

See "Figure 32".

(a) Description

The optional GTS 825 Traffic Advisory System provides real-time traffic information to the PFD (and, indirectly, to the MFD). The Garmin GTS 825 connects with the #1 GDU and communicates with the MFD through an HSDB connection. Power is received from the Avionics Bus #1.

GTS 825 Processor
Figure 32

(b) Troubleshooting

See "Chart 29" on page 342501128.

(c) Removal

1) Remove the baggage closeout panel to gain access to the lower aft avionics shelf. See 25-20-00, Interior Panels - Cabin, Removal.

2) Disconnect the GTS 825 connectors.

3) Loosen the GTS 825 unit by unscrewing the knurled retainers.

4) Move the retainer arms downward. Gently remove the GTS 825 from its rack.

(d) Installation

1) Insert the GTS 825 into the rack and lift each retainer arm to engage lip.

2) Gently screw retainers to secure the GTS 825 unit into place.

3) Visually inspect the connectors and pins for damage. Repair any damage. Connect the connectors to the unit.

4) Reinstall the cabin rear closeout panel. See 25-20-00, Interior Panels - Cabin, Installation.

(e) Return to Service

1) Original GTS 825 Reinstalled

No software or configuration loading is required if the removed GWX 68 is reinstalled. Continue to "GTS 825" on page 342501208.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GTS 825 Installed

If a new, repaired, or exchanged GTS 825 is installed, the correct software and configuration files must be loaded to the unit. See "G1000 Software/Configuration Procedure" on page 342501150 and then continue to "GTS 825" on page 342501208.

CHART 29 (Sheet 1 of 2)
ADVISORY MESSAGES - GTS 825

Advisory Message	Possible Cause	Recommended Actions
CALIBRATION DATA FAULT	Stored factory calibration parameters are invalid.	Replace the unit.
CONFIGURATION DATA FAULT	Stored system configuration parameters are invalid or Mode S address is invalid (All 0’s or F’s). Fault will persist until configuration is corrected.	Reconfiguration the GTS 8XX. If problem persists, replace the unit.
FPGA FAULT	Check of the FPGA image failed. Fault will persist until valid FPGA image is loaded.	If upload of FPGA image was recently attempted, retry the upload. Otherwise, replace the unit.
ROM FAULT	Internal non-volatile memory failure, or invalid data image detected.	If upload of audio image or IGRF magnetic field image was recently attempted, retry the upload. Otherwise replace the unit.
EXECUTION FAULT	CPU execution fault has occurred.	Cycle power and retry self test If fault persists, replace the unit.
ELECTRICAL FAULT	One of the internal electrical voltages are out of range. Fault will persist until power is cycled.	Check aircraft power supply. If fault persists, replace the unit.
WHISPER SHOUT FAULT	Transmitted power is out of tolerance.	Check cable loss configuration, antenna installation and all cable connections and retry self test. If fault persists, replace the unit.
TRANSMIT POWER FAULT	One of the internal transmitter power source voltages are out of range. Fault will persist until power is cycled.	Check aircraft power supply. If fault persists, replace unit.
1030 MHZ FREQUENCY SOURCE FAULT	Transmit Frequency synthesizer is not locked.	Cycle power and retry self test. If fault persists, replace unit.
1090 MHZ FREQUENCY SOURCE FAULT	Receive Frequency synthesizer is not locked.	Cycle power and retry self test. If fault persists, replace unit.
RECEIVER CALIBRATION FAULT	N/A	Check antenna installation and all cable connections and retry self test. Ensure that self test occurs in area free of buildings and large objects that can reflect signals. If fault persists, replace unit.

CHART 29 (Sheet 2 of 2)
ADVISORY MESSAGES - GTS 825

Advisory Message	Possible Cause	Recommended Actions
TRANSMITTER CALIBRATION FAULT	N/A	Check antenna installation and all cable connections and retry self test. Ensure that self test occurs in area free of buildings and large objects that can reflect signals. If fault persists, replace unit.
BAROMETRIC ALTITUDE INPUT FAULT	Own ship barometric altitude calculation is invalid or has timed out.	Check wiring to source of barometric altitude and ensure that source is operating. Fault will clear as soon as valid barometric altitude data is received.
MAIN BOARD TEMPERATURE FAULT	Main board temperature or RF receiver temperature is greater than 90° Celsius or less than -60° Celsius.	Fault will persist until internal temperature returns to acceptable range.
TCAS EQUIPAGE TIMEOUT FAULT	TCAS Equipage data is not being received or has timed out for 800ms.	Check wiring to TCAS Equipage data source and ensure that source is operating. Fault will clear as soon as valid TCAS Equipage data is received.

(20) Configuration Module

See "Figure 33".

The configuration module is located in the backshell of the connector it is terminated to.

(a) Removal

1) Disconnect connector from LRU.

2) Remove 2 screws (8) from cover (7) and remove cover.

3) Unplug connector from configuration module (1).

4) Remove configuration module.

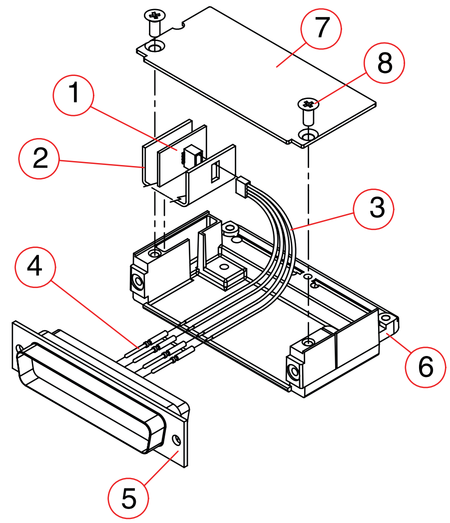
(b) Installation

1) Inspect connector for damaged pins (4).

2) Place configuration module (1) in position.

3) Insert connector into configuration module (1).

4) Assembly of the connector is the reverse of disassembly.



Configuration Module
Figure 33

(c) Return to Service

1) If a GRS 77 AHRS Configuration Module is replaced:

All three GRS 77 / GMU 44 calibration procedures must be performed. Proceed to "GRS 77 / GMU 44 Calibration Procedures" on page 342501190.

2) If a GDC 74A Configuration Module is replaced:

Configuration settings must be reloaded to the GDC 74A. Continue to "G1000 Software/Configuration Procedure" on page 342501150, then perform "GDC 74A Air Data Computer Tests" on page 342501188.

3) If the GEA 71 Configuration Module is replaced:

Perform "GEA 71 Test" on page 342501187.

4) If the Master Configuration Module is replaced:

a) Start the G1000 system in configuration mode.

b) Go to the Configuration Upload Page on the PFD.

c) Press the UPDT CFG softkey.

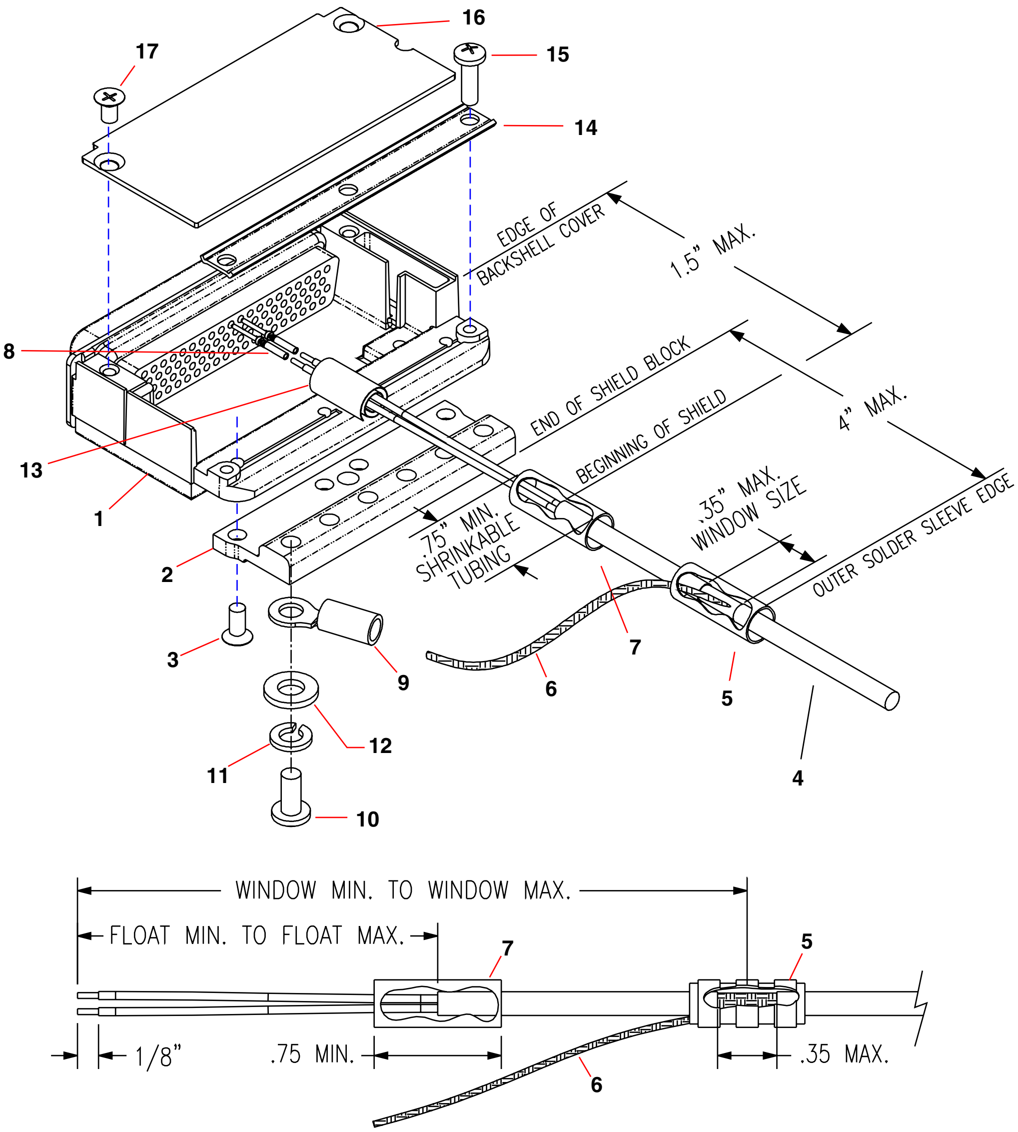
5) If both the PFD and Master Configuration Module is replaced:

The G1000 system (except GRS 77/GMU 44 and GDC 74A) must be reconfigured, continue to "G1000 Software/Configuration Procedure" on page 342501150.

(21) Shield Block Installation

See "Figure 34" on page 342501131.

Most G1000 connectors employ a shield block grounding system to provide necessary ground reference to shielding and/or transducers.



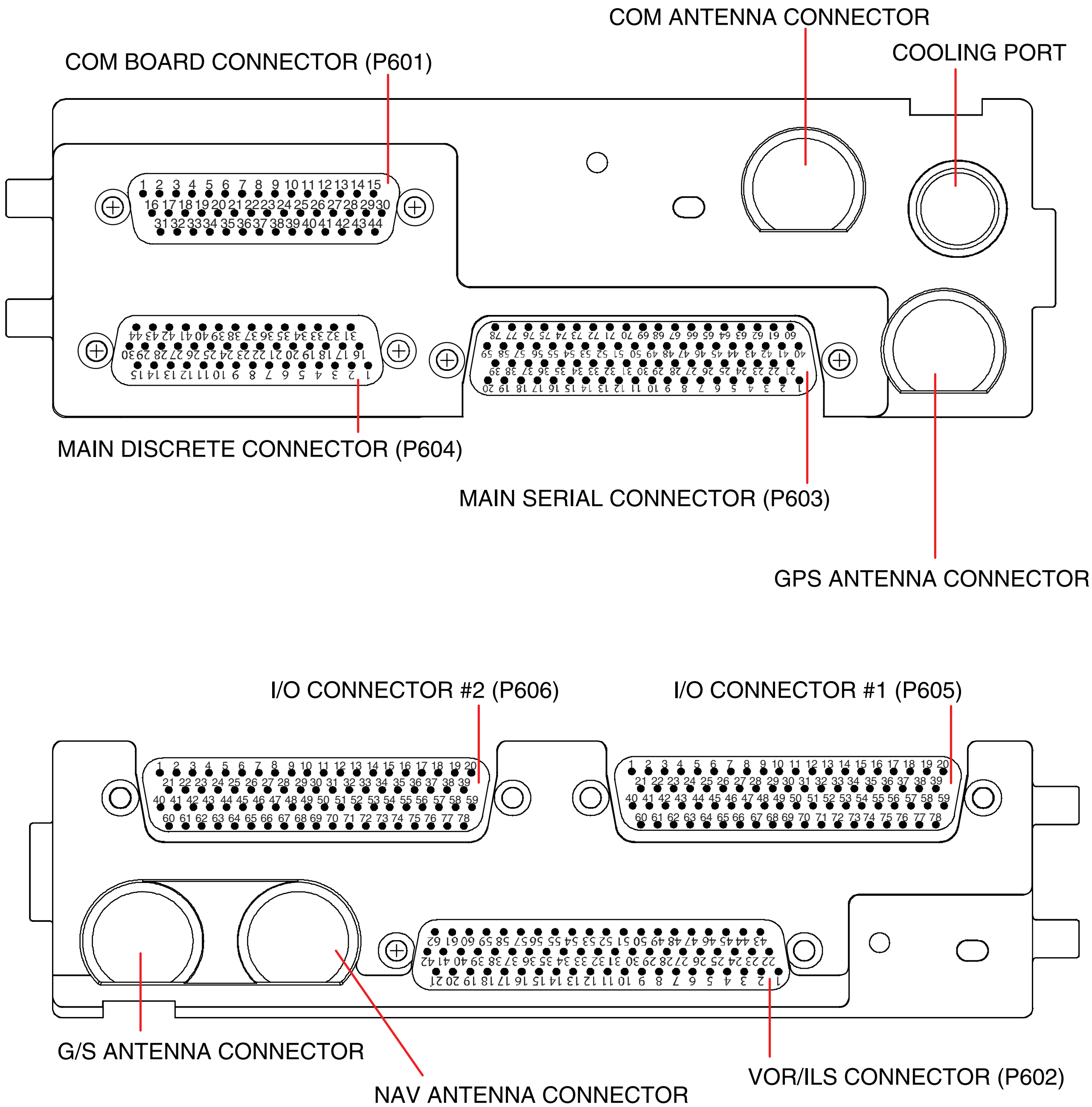
1. CAST HOUSING
2. SHIELD BLOCK(S)
3. SCREW
4. MULTIPLE CONDUCTOR SHIELDED CABLE
   (2–CONDUCTOR DEMONSTRATED HERE)
5. DRAIN WIRE SHIELD TERMINATION (METHOD OPTIONAL)
6. BRAID, FLAT
7. FLOATING SHIELD TERMINATION (METHOD OPTIONAL)
8. PINS
9. RING TERMINAL, #8, INSULATED
10. SCREW
11. SPLIT WASHER
12. FLAT WASHER
13. SILICON FUSION TAPE
14. STRAIN RELIEF
15. SCREW
16. LID
17. SCREW

Shield Block Installation to Backshell
Figure 34

(22) Backshell Connectors

See "Figure 35" on page 342501133.

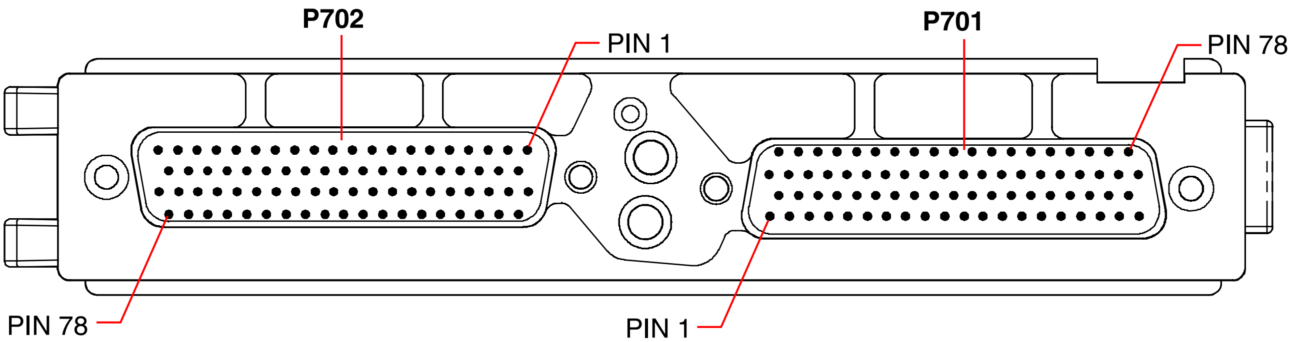
GIA 64(W)



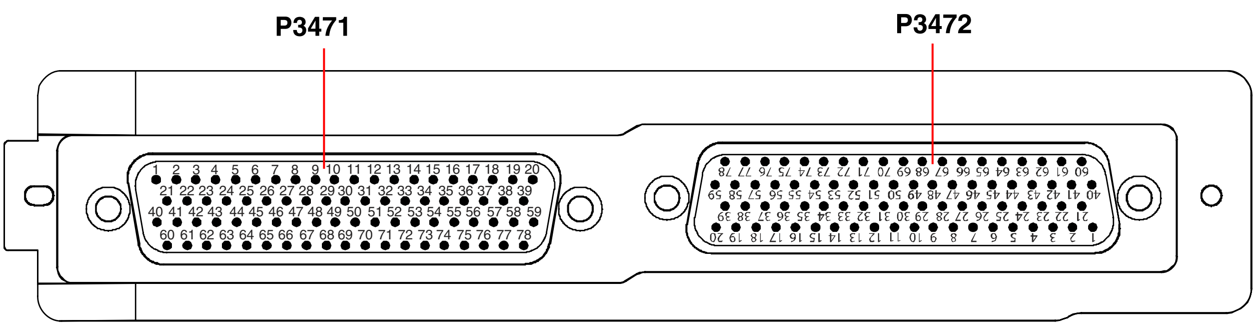
Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 35 (Sheet 1 of 5)

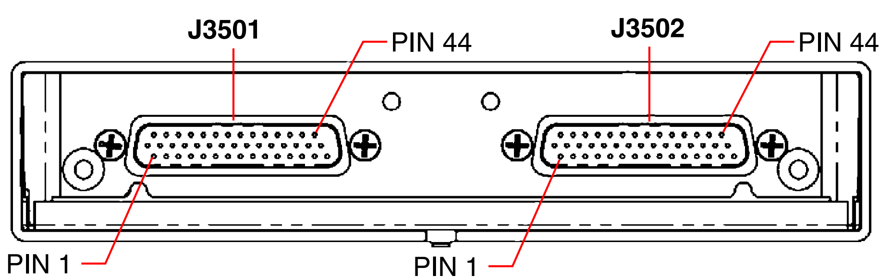
GEA 71



GMA 347



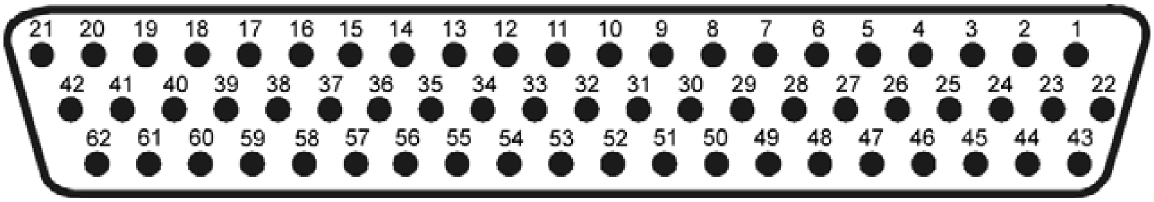
GMA 350



Backshell connectors as viewed with the LRU removed.

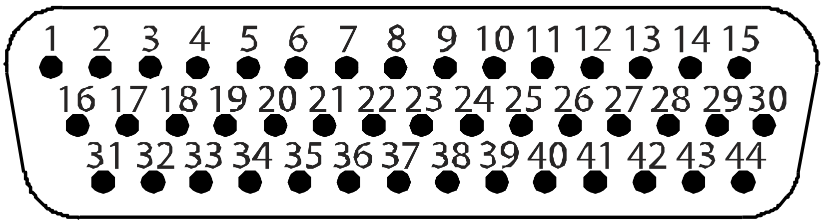
Backshell Connectors
Figure 35 (Sheet 2 of 5)

GDU 1040A / GDU 1500 / GDU 1240A



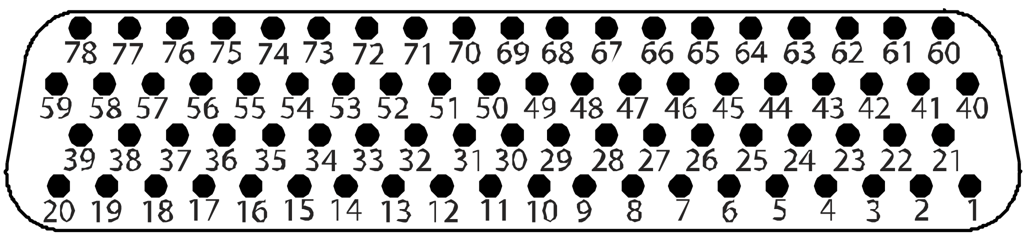
#P10001

GRS 77



#P771

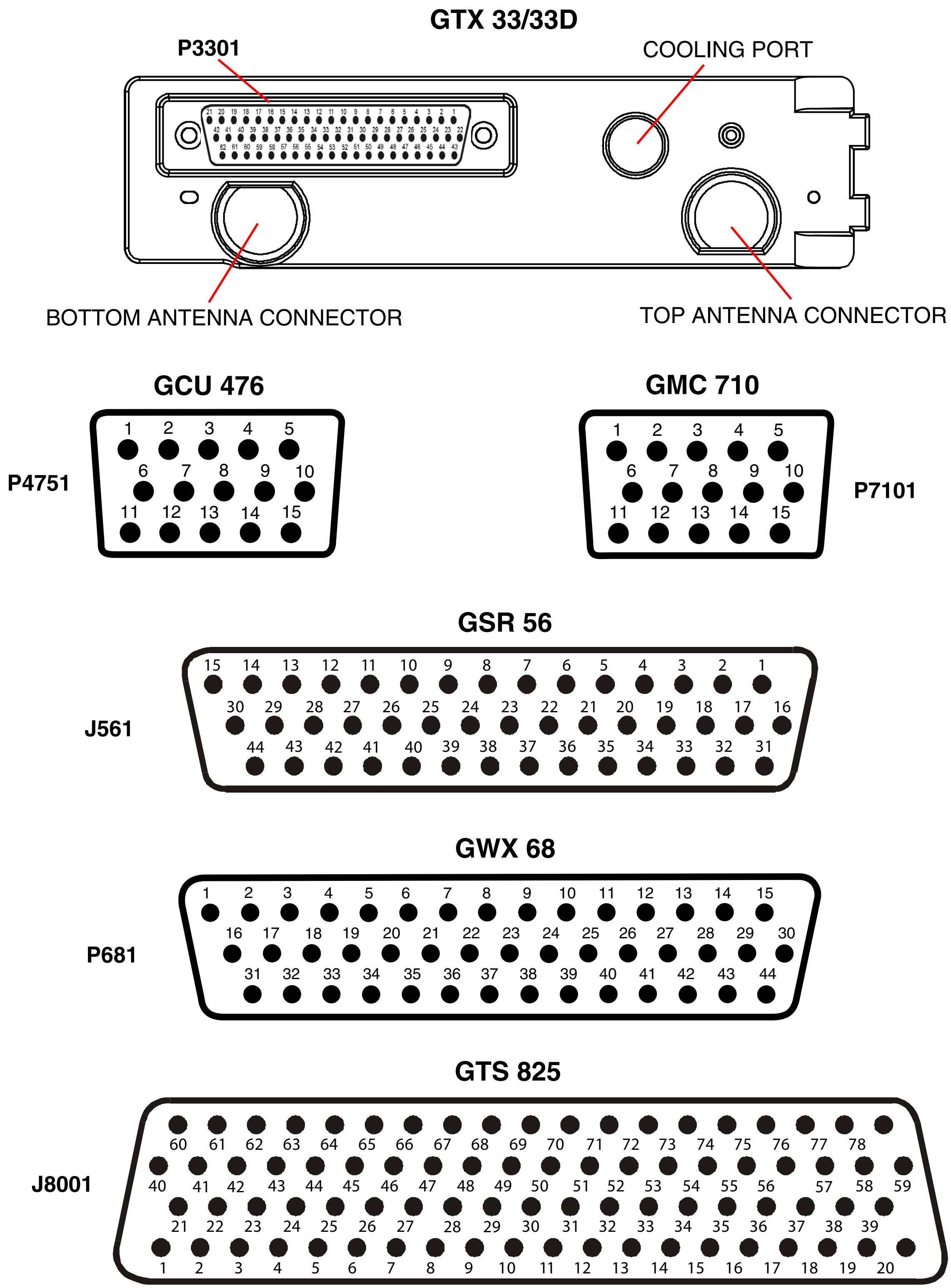
GDC 74A



#P741

Backshell connectors as viewed with the LRU removed.

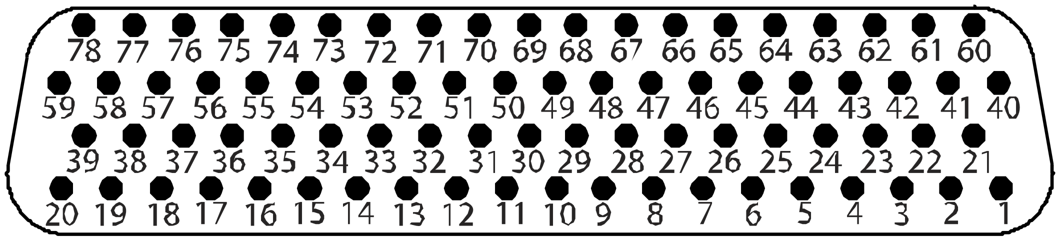
Backshell Connectors
Figure 35 (Sheet 3 of 5)



Backshell connectors as viewed with the LRU removed.

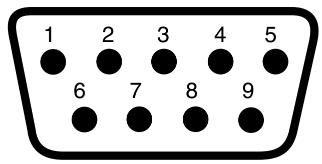
Backshell Connectors
Figure 35 (Sheet 4 of 5)

GDL 69A



P691

GRT 10



P771

Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 35 (Sheet 5 of 5)

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H. Configuration Mode Overview

CAUTION: THE CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

The Configuration Mode exists to provide the avionics technician with a means of configuring, checking, and calibrating various G1000 sub-systems. Troubleshooting and diagnostics information can also be viewed in this mode.

(1) System Configuration and Status Verification

(a) Enter the configuration mode on the PFDs and the MFD by applying power while holding ENT on each display.

(b) Verify on the "SYSTEM STATUS PAGE" that the following are valid. (Green annunciations indicate that the system is healthy): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX.

(c) Rotate the small FMS knob to navigate to the "SYSTEM CONFIGURATION PAGE" and verify that the following are valid. (Indicated by green annunciations): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), and GWX.

NOTE: For a complete description and breakdown of each Configuration Mode page, refer to the G1000 Line Maintenance & Configuration Manual, Garmin P/N 190-00303-04.

(2) SET>ACTV Configuration

CAUTION: THE ACTV>SET SOFTKEY MUST BE USED WITH CAUTION! IF AN IMPROPERLY CONFIGURED UNIT IS INSTALLED, THIS SOFTKEY CAUSES THE WRONG CONFIGURATION TO REPLACE THE CORRECT AIRCRAFT CONFIGURATION.

Throughout the configuration mode pages, there are SET and ACTIVE columns for input/output settings and other parameters.

“SET”: Refers to a setting or group of settings that reside in PFD Internal Memory and/or the Master Configuration Module.

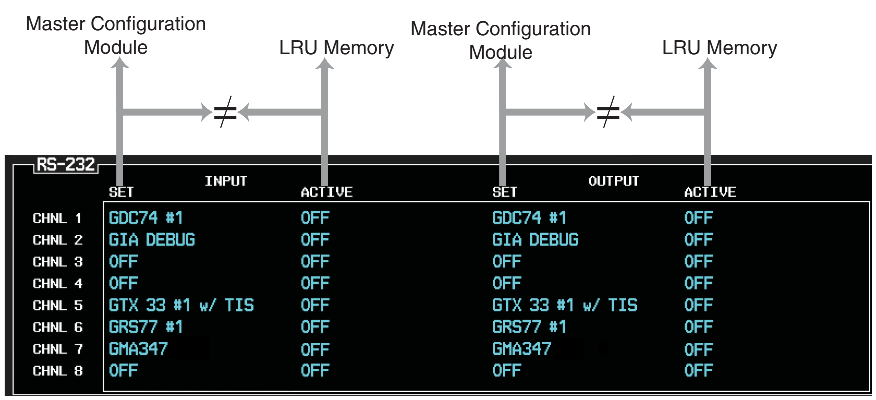
ACTIVE: Refers to an ‘active’ setting or parameter currently being used by the LRU. LRUs store the ‘active’ settings within internal memory.

Data can be manually copied from one column to the other (and consequently from PFD memory to the LRU memory and vice-versa) by using the following two softkeys, when available:

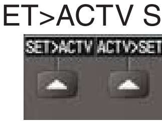
- **SET>ACTV** (read ‘Set to Active’) softkey: Allows the installer to send the information in the SET column (data stored in the master config module) to the ACTV column (data used by LRU).
- **ACTV>SET** (read ‘Active to Set’) softkey: Causes the LRUs current settings to be copied to the master configuration module as SET items.

In the first example shown in "Figure 36", the SET columns do not match the ACTIVE columns. The inequality between SET and ACTIVE indicates a configuration mismatch. By pressing the SET>ACTV softkey, this copies the SET column to the LRU unit's configuration memory. The settings then become the ACTIVE settings for the LRU being configured.

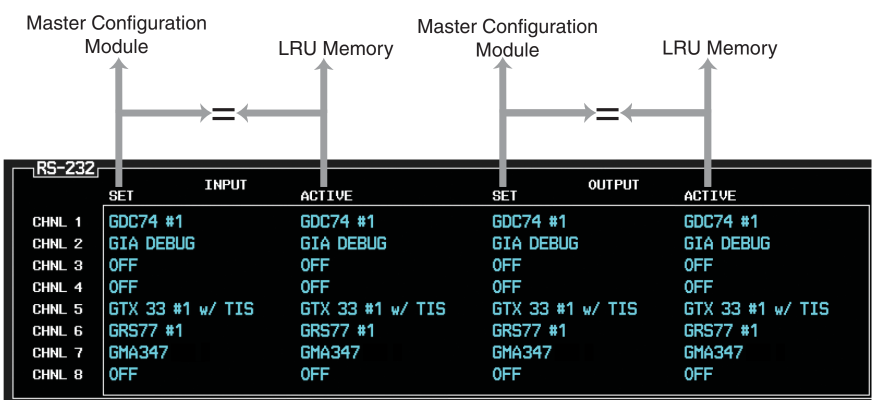
Configuration Mismatch



SET>ACTV Softkey



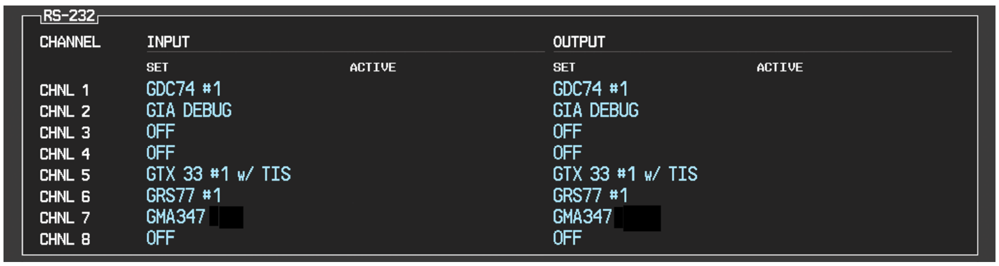
Configuration Correct



SET>ACTV Configuration
Figure 36

When troubleshooting the system, technicians can look for inequalities between SET and ACTIVE columns. Certain problems can be resolved simply by pressing the SET>ACTV softkey, which reloads settings to the specific LRU from the PFD (Note that this can also be accomplished by reloading the configuration files for the LRU, using the software loader card.).

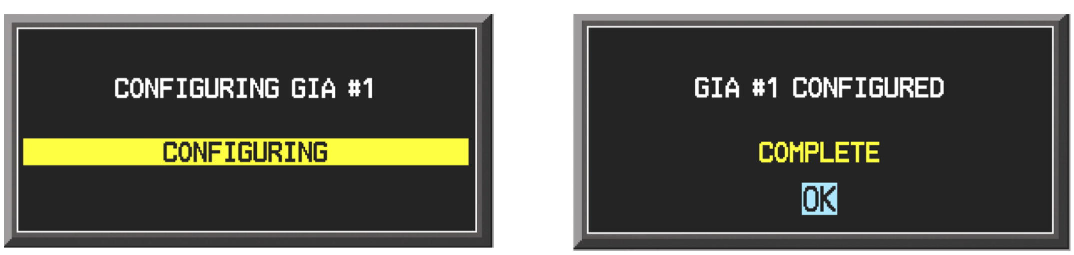
A blank active column, as shown in "Figure 37", represents loss of communication between the display and the particular unit. See troubleshooting for more details.



Blank Active Columns
Figure 37

(3) Configuration Prompts

When configuration settings are changed, the technician receives on-screen prompts and/or confirmations such as those shown in "Figure 38". Other prompts may be encountered during the configuration process.



Configuration Prompts
Figure 38

(4) Data Transmission Indicators

See "Figure 39".

Several configuration screens utilize an indicator light system to show discrete (ON/OFF) data and/or hardware component status. Unless otherwise noted, the following applies to all such status indicators:

(a) Green Light: Expected data is successfully received and is ON. A green light could also indicate that the parameter/component is working correctly.

(b) Red Light: Expected data is not received. A red light could also indicate that a parameter/ component is invalid.

(c) No Light (Black): Expected data is successfully received and is OFF, or no data is expected. A black light could also indicate that the parameter/component is not responding.

(5) Configuration Mode Navigation

Using the FMS knob, a user can navigate through different pages and page groups in the Configuration Mode. See "Chart 30" on page 342501143. For complete description and breakdown of each page, refer to the G1000 Line Maintenance & Configuration manual.

Data Transmission Indicators
Figure 39

CHART 30
CONFIGURATION MODE NAVIGATION

System Page Group		
1. System Status	6. File Manager	11. System Setup
2. Date/Time Setup	7. Diagnostics Terminal	12. Manifest Configuration
3. Main Lighting	8. OEM Diagnostics	13. Maintenance Log
4. Audio Alert Configuration	9. System Configuration	
5. System Upload	10. System Data Path Config	
GDU Page Group		
1. RS-232 / ARINC 429 Config	4. Diagnostics	7. Airframe Configuration
2. GDU Status	5. Serial/Ethernet I/O	8. TAWS Configuration
3. GDU Test	6. Alert Configuration	
GIA Page Group		
1. RS-232 / ARINC 429 Config	3. GIA I/O Configuration	5. GIA Status
2. GIA RS-485 Config	4. COM Setup	6. GIA CAN Config
GEA Page Group		
1. Engine Data	2. GEA Status	3. GEA Configuration
GTX Page Group		
1. RS-232 / ARINC 429 Config	2. Transponder Configuration	
GRS Page Group		
1. AHRS / Air Data Input	2. GRS / GMU Calibration	
ADC Page Group		
1. ADC Configuration	2. GDC Configuration	
GFC Page Group		
1. GFC Configuration	2. GFC Status	
GMA Page Group		
1. GMA Configuration		
GDL Page Group		
1. GDL 69 Config		
RMT Page Group		
1. Remote Controller Status		
GWX Page Group		
1. GWX Configuration		
OTHER Page Group (Group not present unless Stormscope configured ON.)		
1. Stormscope		
CAL Page Group		
1. Fuel Temperature Calibration	2. Flaps and Trim Calibration	3. HSCM Calibration

I. G1000 / GFC 700 Software Information

NOTE: The following sections provide a detailed description of loading all G1000 software and configuration files, which may be excessive for individual LRU removal and replacement. If removing and replacing individual LRUs, refer to Components, above, for the necessary steps.

(1) G1000 Software Image

CAUTION: IT IS CRITICAL THAT THE TECHNICIAN INSTALLS THE CORRECT SOFTWARE IMAGE PART NUMBER WHEN SERVICING THE G1000 SYSTEM. APPROVED SOFTWARE DEFINITIONS FOR EACH CONFIGURATION ARE DEFINED IN THE REQUIRED EQUIPMENT LIST (REL), "CHART 39" ON PAGE 342501229.

All software and configuration files were certified by Garmin as part of the FAA-approved Type Design data. Approved software definitions for each configuration are defined in the appropriate Required Equipment List (REL), "Chart 39" on page 342501229.

G1000 software and configuration files are controlled via the approved software image part number listed in the REL. This software image is loaded into the G1000 using a software loader card. The installer shall create this software loader card by downloading the approved software image in accordance with the procedure provided immediately following.

CAUTION: BE CAUTIOUS WHEN USING SOFTWARE LOADER CARDS DURING MAINTENANCE. THE G1000 SYSTEM IMMEDIATELY INITIALIZES THE CARD UPON POWER-UP. ON-SCREEN PROMPTS MUST BE GIVEN CAREFUL ATTENTION IN ORDER TO AVOID POTENTIAL LOSS OF DATA. ALWAYS READ THROUGH PROCEDURES GIVEN IN THIS SECTION, BEFORE ATTEMPTING TO USE THE SOFTWARE LOADER CARDS.

(2) Software Loader Card Creation

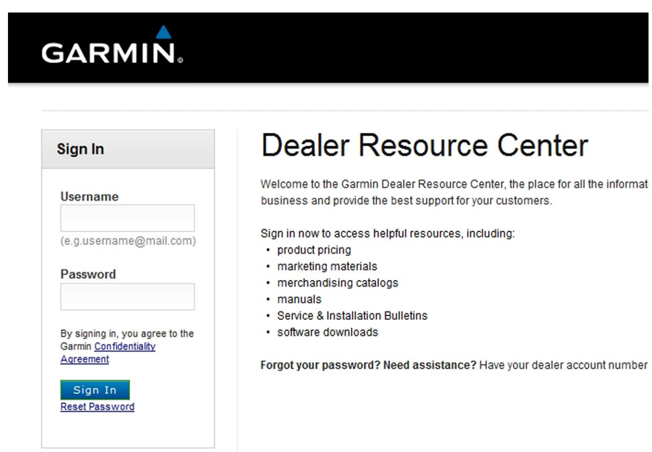
The following procedure requires a PC with an SD card reader. Piper recommends SanDisk® SDDR-99 and SDDR-93 card readers. Additionally, only SanDisk brand 16GB SD cards are recommended.

The software image is an executable self-extracting file which builds the correct file structure onto an SD card for use loading software to the G1000 and GFC700. To obtain the current file follow the procedures outlined below.

NOTE: In order to create a loader card, the installer completing these procedures must be an authorized Piper service center to gain access to the necessary data via the Garmin website.

(a) Go to http://www.garmin.com and click on the 'Dealer Resource Center' link in the lower right hand portion of the home page. Enter username and password. See "Figure 40".

(b) Select Support and then Software Downloads.



Dealer Login Screen
Figure 40

(c) Search for the appropriate image version (i.e., "0720.xx" from the appropriate "Chart 39" on page 342501229) in the “Search by Keyword” field.

(d) Save the file to a known location on the PC.

(e) Run the executable file that was downloaded. A window will appear to confirm software installation.

(f) Insert the SD card into the SD card reader and select the Next button in the pop-up window.

(g) In the pop-up window, select the drive that corresponds to the inserted SD card and then click the Next button.

NOTE: The selected drive will be completely erased. Click the Next button to acknowledge any warnings that appear. Another pop-up window will appear to monitor loader card creation progress. After the card creation is successfully completed, another pop-up window will appear.

(h) Select the Finish button.

(i) Stop the card reader using Windows stop device function.

(j) Eject the SD card from the card reader.

(k) Apply a label to the software loader card identifying the aircraft, G1000 SFTW and the appropriate software image part number as shown below:

NOTE: See appropriate "Chart 41" on page 342502227 for software image part number.

PA-46-350P
G1000 SFTW
006-x0720-xx

(3) Handling a Software Download Card

The SD memory cards used for the Software Download Card are manufactured to the specifications of the SD Card Association for use in portable devices. As such they are reasonably robust devices that do not require special handling instructions. However, the memory cards are not unbreakable and reasonable precautions regarding their use and handling should be used including the following guidelines:

(a) Do no bend an SD memory card or subject it to violent impact.

(b) Do not leave SD cards in locations where they may be exposed to moisture or high temperatures.

(c) Avoid exposing the card to static electricity or electro-magnetic interference.

J. Software Files

(1) Definition

Software files are defined by part number and version number. Each G1000 / GFC 700 LRU reports the software version it currently contains to the user in two places.

(a) Normal System Mode: The AUX – SYSTEM STATUS page lists each LRU and the reported software version.

(b) Configuration Mode: The SYSTEM STATUS page (SYSTEM page group) reports more detailed LRU information, including software version, part number, and LRU status. Software files are loaded to LRUs from the SYSTEM UPLOAD page in configuration mode.

(2) Troubleshooting

See "Chart 31" on page 342501147.

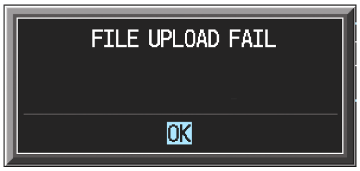
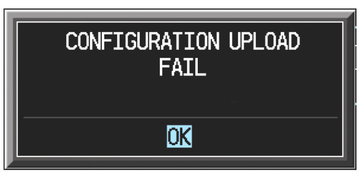
(3) Configuration File Descriptions

(a) Configuration Files are divided into groups, and are only displayed at the System Upload page during the software/configuration loading process.

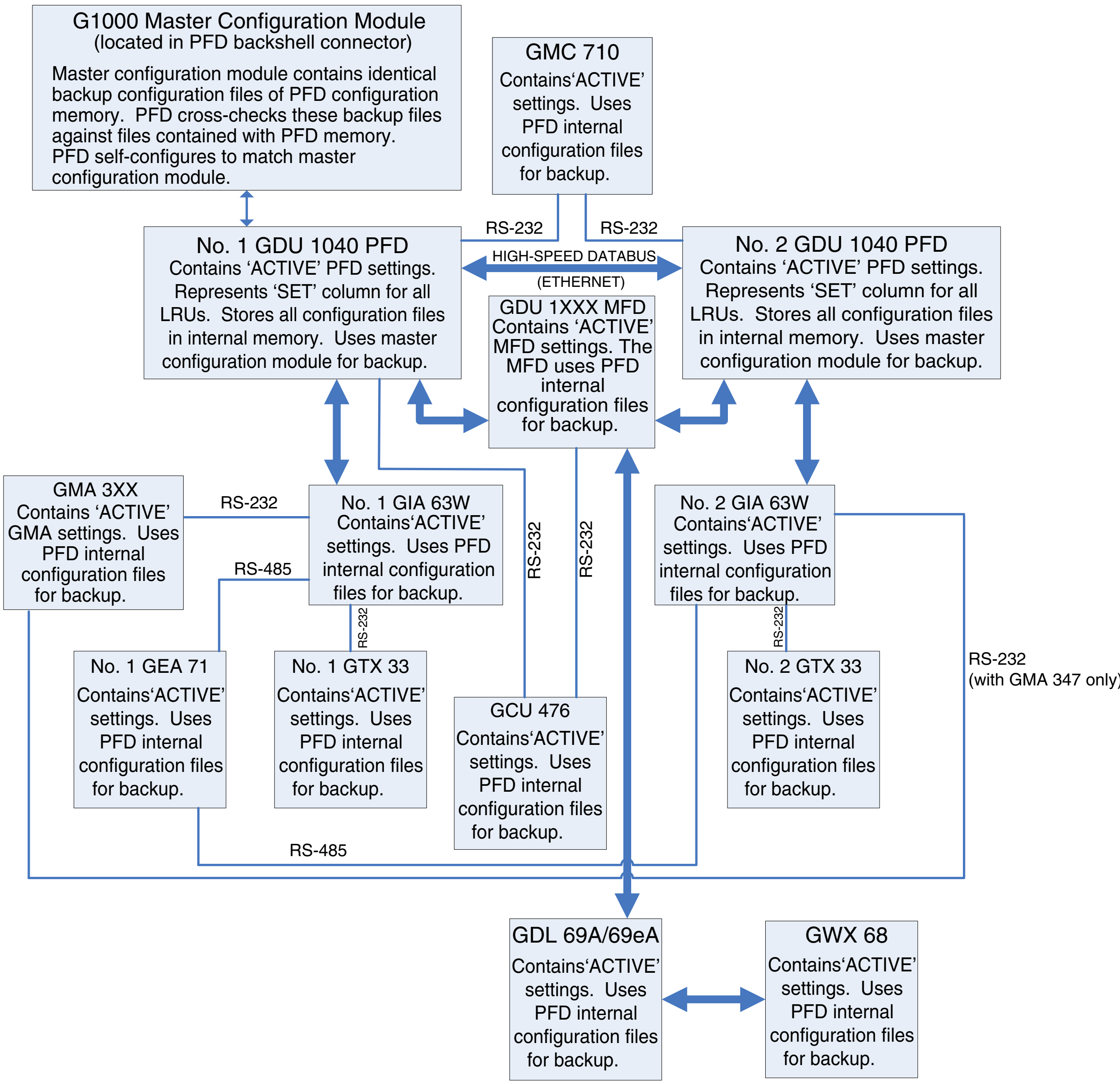
(b) Configuration files contain preset selections for input/output channels, aircraft-specific settings, and LRU-specific settings. The following list describes each of the configuration files:

AIRFRAME	This file contains data such as airspeed parameters, engine/airframe sensor limitations, fuel tank parameters and alerting system settings that tailor a PFD or MFD to the PA-46-350P or PA-46R-350T.
ALERTS	Specific alerts are set when this file is loaded.
SYSTEM	This file configures the G1000 high-speed data bus (HSDB) to expect two PFDs, MFD, two GIAs, and a GDL69A/69eA.
MANIFEST	This file loads a manifest checklist of all software part numbers and versions associated with an approved system configuration. The G1000 performs a software check between each LRU's reported version and the version contained on the manifest. If an inequality is detected for an LRU, this LRU is then excluded from the G1000 and a manifest alert is triggered to the operator.
MFD1	This file configures MFD serial/discrete communication and alert system settings.
PFD1/PFD2	These files configure PFD1/PFD2 serial/discrete communication and alert system settings.
GCU	This file configures GCU 476 serial/discrete communication settings.
GMC	This file configures GMC 710 serial/discrete communication settings.
GIA1/GIA2	These files configure GIA1/GIA2 serial/discrete communication settings.
GMA_PIL/	This file configures GMA audio and serial communication settings.
GTX1/GTX2	These files configure GTX 1/GTX 2 transponder and serial communications settings.
GEA	These files configure GEA engine/airframe parameters.
GDC_PIL/ GDC_COPIL	This file configures GDC 74A air data values.
GDL_69	This file configures gains and cable loss.
GWX	This file configures parameters for the radar installation.
AUDIO	This file configures audio alerts.

CHART 31
G1000 - TROUBLESHOOTING / MESSAGE ADVISORIES - SOFTWARE CONFIGURATION

Problem	Solutions
MFD or PFD displays do not power up:	• Ensure power is present at display backshell connector. • Troubleshoot per the “Blank Display” GDU section.
Software file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry software file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with software version and Card Loader. • Replace LRU.
Configuration file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry configuration file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with Card Loader. • Replace LRU.
GIA1 and/or GIA2 to ‘LRU’ data path not working:	• Ensure GIA1 and GIA2 are configured correctly. • Check wiring, connectors & pins as needed.
Software File Mismatch Alert appears in lower right corner of PFD when started in normal mode: 	• Ensure that proper software file part number and version were loaded to LRU. See the appropriate Required Equipment List (REL), "Chart 39" on page 342501229. • Check and ensure that correct Loader Card was used during load process. (1) • Reload software to LRU.

(1) "G1000 / GFC 700 Software Information" on page 342501144



LRU Configuration File Storage
Figure 41

(4) Configuration File Storage

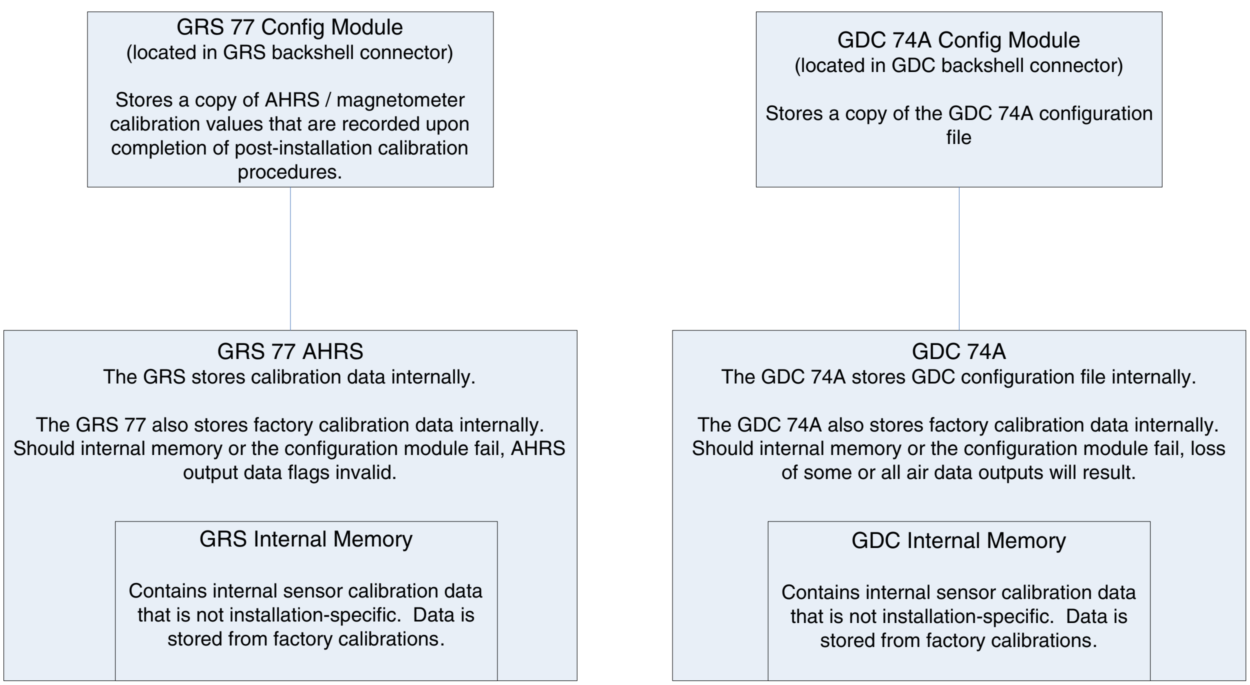
The G1000 system is designed to store all configuration settings in various places so that the configuration is retained in the aircraft during maintenance of units.

During system configuration, as shown in "Figure 41" on page 342501148, each file is sent directly to the applicable LRU where it is stored in local LRU memory (except GRS 77 & GDC 74A). Each file is also stored in the PFD internal memory. The applicable PFD also sends a copy of all configuration files to the 'Master Configuration module', located in the connector backshell (see "Figure 33" on page 342501130). If the PFD is replaced, the configuration module retains all configuration files in the aircraft.

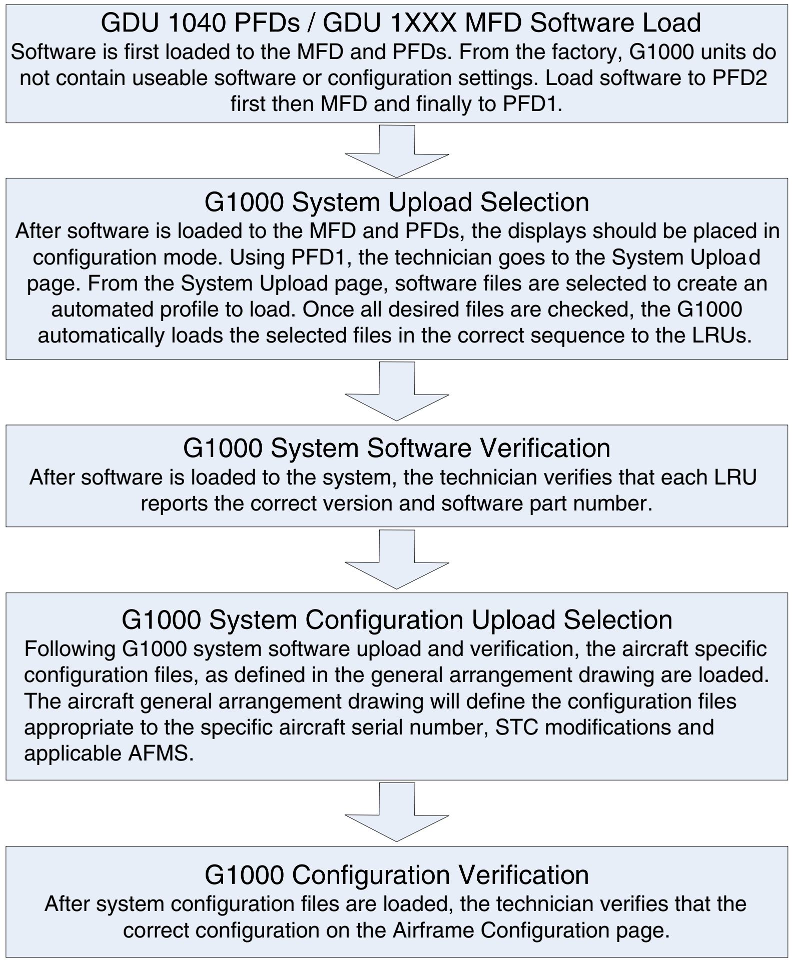
NOTE: The GRS 77 AHRS and GMU 44 Magnetometer do not have a configuration file. However, these LRUs do store calibration data acquired during the post installation checkout, which are characteristic to the specific installation. While performing maintenance on these units, re-calibration may be required. See GRS 77 AHRS / GMU 44 Magnetometer for more information on re-calibration criteria.

The GRS 77 and GDC 74A configuration modules, as shown in "Figure 42", function differently than the rest of the system. The GDC 74A's configuration file is loaded directly to GDC internal memory. A copy of the file is stored in the GDC configuration module.

The GRS 77 configuration module does not store any configuration settings. Instead, it stores calibration data recorded during installation calibration procedures.



GRS/GDC Configuration Settings Storage
Figure 42



Software/Configuration Overview
Figure 43

(5) G1000 Software/Configuration Procedure

See "Chart 32" on page 342501151.

This section summarizes the procedures required to load software and configuration files to the G1000. It is intended to work as a central guide for technicians to use while performing maintenance on the aircraft. In sections of this manual where software is required to be reloaded, these sections will make reference back to this Section for instructions. The technician should use proper judgment regarding the context of maintenance required while following this section.

The diagram in "Figure 43" depicts an overview of the software/configuration sequence for the G1000 system. This applies mostly to a new G1000 system which has not previously been powered up and is for informative purposes only.

CHART 32
G1000 - SYSTEM COMMUNICATION HIERARCHY

The following criteria must be satisfied to be able to perform the desired operation:	
Desired Operation	**Criteria for Success**
Load Software to MFD or PFD Displays	• G1000 Loader Card must be inserted in top slot for each display to be loaded. • CLR & ENT keys must be held during power up of display. • Power on only one display at a time during software loading.
Load AIRFRAME, SYSTEM, MFD, PFD1, PFD2 and MANIFEST configuration files to MFD and PFDs	• G1000 Loader Card must be inserted in top slot of PFD1. • PFD1 and MFD must be powered on. • PFD1 and MFD must have correct software.
Load Software/Configuration files to GIA 63Ws	• G1000 Loader Card must be inserted in top slot of PFD1. • G1000 system must be powered on. • PFD and MFD must have correct software. • PFD 1 and MFD must be successfully configured with AIRFRAME, SYSTEM, MANIFEST, MFD, PFD1 and PFD2 configuration files.
Load Software/Configuration files to: - GMA 347 / GMA 350 - GDC 74A - GEA 71 - GRS 77 (software only) - GMU 44 (software only) - GTX 33	• G1000 Loader Card must be inserted into PFD1 top slot. • G1000 must be powered on. • PFD1 and MFD must have correct software and configuration settings. • GIA 63Ws must have correct software. • GIA 63Ws must be successfully configured with GIA1 and GIA2 configuration files. • Data path from GIA1 to each LRU must be operational.

K. Software Load

(PIR-PPS55026, Rev. L. / 107977 G.)

(1) MFD & PFD Software Load

CAUTION: DO NOT INTERRUPT POWER TO THE AIRCRAFT AT ANY TIME DURING THESE PROCEDURES. INTERRUPTION OF POWER COULD RESULT IN DAMAGE TO THE EQUIPMENT.

(a) Apply power to the aircraft and all avionics.

(b) Pull the MFD and both PFD circuit breakers. Remove any SD cards that were previously inserted in either PFD or MFD.

(c) Insert G1000/PA-46 Loader Card into the PFD2 top card slot.

(d) While holding the ENT key on the PFD2, restore power to the PFD2 by pushing in the PFD2 circuit breaker.

NOTE: For loading MFD, the far right softkey can be used in place of the ENT key of the PFD2 and PFD1.

(e) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFD2, release the ENT key.

(f) Press the YES softkey at the following prompt (prompt may not appear):

“DO YOU WANT TO CLEAR USER SETTINGS?”

“YES WILL BE ASSUMED IN 10 SECONDS”

(g) Press the YES softkey at the following prompt:

“DO YOU WANT TO UPDATE SYSTEM FILES?”

“NO WILL BE ASSUMED IN 30 SECONDS”

“UPDATING SYSTEM FILES, PLEASE WAIT.”

(h) Press the YES softkey at the following prompt (prompt may not appear):

“DO YOU WANT TO UPDATE SPLASH SCREEN?” (i.e., Custom Graphics Files)

“NO WILL BE ASSUMED IN 30 SECONDS”

(i) Press any softkey at the following prompt to confirm load completion:

“PRESS ANY KEY TO CONTINUE.”

“CONTINUING IN 10 SECONDS.”

When loading PFD2, go to step (j). For MFD, go to step (l). For PFD1, go to step (m).

(j) New software is loaded to the PFD2. When complete, the PFD2 starts in configuration mode.

(k) Remove the G1000/PA-46 Loader Card from PFD2 and insert it into the top card slot on MFD and repeat steps (d) thru (i). The rightmost soft key is used in place of the ENT key.

(l) Remove the G1000/PA-46 Loader Card from the MFD and insert it into the top card slot on PFD1 and repeat steps (d) thru (i). The ENT key is used in place of the softkey.

(m) When PFD1 update is complete, it starts in the configuration mode. DO NOT remove power.

CAUTION: FOR THE REST OF THE SOFTWARE/CONFIGURATION PROCEDURE, DO NOT OPERATE THE MFD OR PFD2 WHILE LOADING SOFTWARE OR CONFIGURATION FILES UNLESS SPECIFICALLY INSTRUCTED TO DO SO. A FAILED OR CANCELLED LOAD MAY RESULT.

(2) System Upload

NOTE: To Select/Deselect the cursor in fields, press the small FMS knob on the PFD. The large FMS knob changes Groups, and the small FMS knob changes Pages within a Group.

(a) Go to the SYSTEM UPLOAD page on PFD1 using the small FMS knob.

(b) S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

1) Activate the cursor, select the Airframe field, and rotate the small FMS knob to generate a pick list. Rotate the small FMS knob to select PA-46-350P or PA-46R-350T.

2) Press ENT.

NOTE: Ensure that the correct airframe type is selected before proceeding; otherwise incorrect configuration information will be loaded. Once an airframe type is selected, the cursor moves to the File field.

3) Rotate the small FMS knob to generate a pick list and select the appropriate aircraft base configuration file.

(c) S/N's 4636633, 4636652 and up.

1) Activate the cursor, select the GROUP field, and rotate the small FMS knob to generate a pick list. Rotate the small FMS knob to select Baseline.

2) Press ENT.

3) In the ITEM field, rotate the small FMS knob to generate a pick list and select the appropriate item = PA-46-350P.

(d) Press ENT.

(e) Press the CHK ALL softkey to select all loadable software and configuration files.

(f) Press the LOAD softkey (The load takes approximately 1.5–2 hours to complete).

(g) Press ENT to accept the software load.

(h) Deselect cursor.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the loader card from top slot of PFD1.

(3) Optional Equipment Configuration

Perform the following steps if any of the following options are installed:

- In S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153

KTA 870 TAS	GWX 68	Dual GTX 33
KR 87 ADF	KN 63 DME	Single GTX 33 ES
WX-500 Stormscope	Vacuum (Matrix only)	Dual GTX 33 ES
GDL 69A (XM)	Stall Heat ON	

NOTE: Stall Heat ON option must be installed in all airplanes.

- In S/N's 4636633, 4636652 and up.

ADF RA-3504	Aft AV Fan Installation	GWX 68
Deice Installation	KN 63 DME	WX-500 Stormscope
Enhanced AFCS Support	ARTEX ELT	SpeedBrake Installation
GDL 69A/69E SXM	GSR 56 Iridium	GSR 56 Voice/Text Only
GTS 825	No. 2 GTX 33 ES	

NOTE: Fields / selections in [brackets] below apply to S/N's 4636633, 4636652 and up.

(a) On PFD1, select the System Upload page and activate cursor.

(b) In the Airframe [GROUP] field, create a pick list by rotating the small FMS knob and select PA-46-350P or PA-46R-350T [Options].

(c) Press ENT.

(d) In the File [ITEM] field, create a pick list by rotating the small FMS knob and select the desired option, press ENT.

(e) Verify both Software and Configuration boxes are checked for Garmin options.

(f) Verify Configuration boxes checked for all other options.

(g) Press LOAD softkey.

(h) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(i) Press ENT to accept.

(j) For other installed options, repeat steps (d) thru (i) by rotating the large FMS knob and highlighting desired option in the File [ITEM] Field.

(k) De-activate the cursor.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the loader card from top slot of PFD1.

(4) Calibration Option Load

NOTE: Only load configuration option once. Re-loading of configuration option could result in a loss of system calibrations.

(a) Go to the System Upload page and activate cursor.

(b) S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153

1) In the Airframe field, create a pick list by rotating the small FMS knob and select PA-46-350P or PA-46R-350T.

2) Press ENT.

3) In the File field, create a pick list by rotating the small FMS knob and select PA-46 Calibration Option.

4) Press LOAD softkey.

5) Verify the Summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(c) S/N's 4636633, 4636652 and up

1) In the GROUP field, create a pick list by rotating the small FMS knob and select Options.

2) Press ENT.

3) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired calibration (Flaps Calibration Initialization or Fuel Calibration Initialization).

4) Press LOAD softkey.

5) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

6) Press UPDT CFG softkey.

7) Press ENT to select YES when prompted "Update Config Module?"

(d) Press ENT to confirm completion.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the loader card from top slot of PFD1.

(5) Splash Screen Update

(a) Remove power from PFD1, PFD2, and MFD.

(b) Insert G1000/PA-46 Loader Card into the PFD2 top card slot.

(c) Apply power to PFD2.

(d) Press the NO softkey at the following prompt:

"DO YOU WANT TO UPDATE SYSTEM FILES?"

(e) Press the YES softkey at the following prompt:

"DO YOU WANT TO UPDATE SPLASH SCREEN?" (i.e., Custom Graphics Files)

(f) At the prompt, press any key to continue.

(g) Remove G1000/PA-46 Loader Card from the PFD2 top card slot.

(h) Repeat steps (b) thru (g) for MFD, and then PFD1.

(i) Upon completion of splash screen update, remove the Loader card from top slot of PFD1.

(6) Software Load Confirmation

(a) Each PFD and MFD running in configuration mode.

(b) On PFD1, select the Systems Status page. Activate the cursor and toggle to the LRU window.

(c) Highlight each of the listed items in the LRU window and check the LRU Part Number, software part number and software version against the current numbers and versions shown in the appropriate Required Equipment List (REL), "Chart 39" on page 342501229. Use "Chart 38" on page 342501228 to record. Some items may be within the expandable list (denoted by a "+" next to the LRU) of the corresponding item. To expand the list, press ENT with the desired LRU highlighted

NOTE: Not all of the software P/Ns and versions on the Systems Status page need to be verified, only those recorded in the appropriate REL.

(d) De-activate the cursor.

NOTE: If any software is not successfully loaded, do not continue with post installation procedures. Troubleshoot and resolve the issue before continuing.

(e) Re-start PFD1, PFD2 and MFD.

(f) On MFD splash screen, verify appropriate version number ("Piper PA-46 System xxxx.xx") is displayed in the upper right corner of the MFD.

(g) Select softkey on MFD to proceed beyond splash screen.

(h) Verify that no manifest (software mismatch) alerts are shown on any of the displays.

L. Configuration and Setup

Start #1 PFD, MFD, and #2 PFD in configuration mode by holding ENT key while applying power.

(1) XM Radio (GDL 69A/69eA)

See "Chart 25" on page 342501115, if the XM Satellite Radio is installed and requires activation.

(2) GRC 10 XM Pairing

If Installed.

NOTE: To pair the GRC 10 XM Handheld Remote Control to the GRT 10 Remote Transceiver, you will need the GRT 10 Serial Number, and the GRC 10 will need to have two AA batteries installed.

(a) Press and Release any key on the GRC 10 to power on the unit.

(b) The GRC 10 will display: "GRT 10 not found"

(c) Press the following buttons in sequential order:

1) UP
2) DOWN
3) LEFT
4) RIGHT
5) MINUS (-)
6) PLUS (+)
7) PSET

(d) Using the arrow buttons on the GRC 10, enter the GRT 10 Serial Number.

(e) Verify that the correct Serial Number was entered.

(f) Store the GRT 10 Serial Number by pressing the XM button.

(3) Aircraft Registration Number Entry (Transponder Configuration)

NOTE: The large FMS knob changes Groups and the small FMS knob changes Pages within a Group.

(a) S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153

1) Select the GTX group, then select the TRANSPONDER CONFIGURATION page on PFD #1.
2) Ensure that the ADDRESS TYPE is US TAIL under the SET and ACTIVE columns.
3) Ensure that the FLIGHT ID TYPE is SAME AS TAIL under the SET and ACTIVE columns.
4) Activate the cursor and highlight the ADDRESS field. Use the small/large FMS knob to enter the aircraft registration number.
5) Once the correct registration number is entered, press the ENT key to configure the transponder.
6) Observe the status window and press the ENT key on PFD #1 to acknowledge prompt.
7) If aircraft has the dual GTX 33 option, use the cursor to select XPDR2 and repeat steps (1) – (6).

(b) S/N's 4636633, 4636652 and up

1) Select the GTX group by rotating the large FMS knob on #1 PFD, and then select the TRANSPONDER CONFIGURATION page by rotating the small FMS knob.
2) Ensure that the ADDRESS TYPE is US TAIL under the SET and ACTIVE columns.
3) Ensure that FLIGHT ID TYPE is SAME AS TAIL under the SET and ACTIVE columns.
4) Select the SYSTEM group by rotating the large FMS knob, and then select the AIRCRAFT CONFIGURATION page by rotating the small FMS knob.
5) Activate the cursor and highlight the AIRCRAFT REGISTRATION field. Use the small/large FMS knob to enter the aircraft registration number.
6) Once the correct registration number is entered, press ENT key. Then press the SET GTX1 softkey to configure the transponder. Press ENT to select OK at the dialog that pops up. Then press ENT again after configuration is completed.
7) If dual GTX are installed, press the SET GTX2 softkey. Press ENT to select OK at the dialog that pops up. Then press ENT again after configuration is completed.
8) If GTS 825 is installed, press SET GTS softkey to configure traffic system. Press ENT to select OK at the dialog that pops up. Then press ENT again after configuration is completed.

(4) Flaps Calibration

NOTE: The large ‘FMS’ knob changes Groups and the small ‘FMS’ knob changes Pages within a Group.

(a) Start the #1 PFD, MFD, and #2 PFD in the configuration mode by holding the ENT key while applying power.

(b) Select the ‘CAL’ group, then select the ‘FLAPS AND TRIM CALIBRATION’ page on ‘PFD1.

(c) Unlock the Calibration page by entering the following softkey password:

- 1 (far left softkey)
- 2
- 3
- 4

(d) Press the ‘FLAP POSITION RESET’ SOFTKEY. Press the ‘ENT’ key to acknowledge the prompt after the ‘RESET’ softkey is pressed.

(e) Put the flaps in the ‘FULL UP’ position.

(f) Press the ‘FLAP POSITION UP’ softkey.

(g) Put the flaps in the ‘FULL DOWN’ position.

(h) Press the ‘FLAP POSITION DN’ softkey.

(i) Return the flaps to the up position.

(j) Verify ‘CAL Value’ in the ‘FLAP POSITION’ window indicates 0.00000 ± 0.02000. If it does not, repeat steps (c) – (h) until it does.

(k) The calibration is complete.

(5) Iridium (GSR 56)

See "Chart 27" on page 342501119, if the Iridium Satellite Radio/Datalink is installed and requires activation.

(6) KTA-870 Configuration

If installed, in S/N's 4636460, 4636463–4636651, less 4636481 and 4636633; or S/N's 4692134 and up, less 4692141, 4692149, 4692153.

(a) Set up computer and connect computer to the TDIAG interconnect on the aircraft to the Com 1 port on the laptop.

NOTE: The "TDIAG" connector is in the harness connected to the lower right side of the KTA-810 Processor installed on the left side of the lower aft equipment shelf.

(b) Start the TAS DIAG software on the PC.

(c) Select "Config Mod" menu item.

(d) Select "Initialize Config Mod".

(e) When the "Initialize Configuration Module" dialogue box appear select "Yes".

(f) When the "Configuration Module" is available select "OK"

(g) When the "Modify Config Module Strap Data" dialogue box appears the "Straps Page 1" tab will be presented in the dialogue box. Make the following selections and then proceed to "Straps Page 2":

Straps Page 1	**Selection**
Heading Type	ARINC 429 Data
Attitude Type	Not Installed
Controller Type	Discrete
Baro Altimeter Type	ARINC 706/429
Airborne Headphones	Max
Ground Headphones	Max
Radar Altimeter Type	Not installed
Straps Page 2	**Selection**
Intruder File Protocol	ARINC 735a
Airborne Functional Test	Allow
Display Intruder Limit	30 Intruders
Heading Sense Valid	Ignore
On-Ground Display Mode	TA Only
Display Valid Input Usage	Ignore
ATE Outputs	Disable
RS-422 Outputs	Disable

(h) When "Straps Page 2" is completed select "OK".

(i) When asked to write data to the config module Select "OK" and TAS will reboot.

(j) After the IHAS reboots select "OK". The Traffic Configuration is complete.

(7) Option Unlock Procedures

WARNING: PRIOR TO ATTEMPTING UNLOCK OF ANY OF THE FOLLOWING OPTIONS, ENSURE THAT THE SOFTWARE IS PROPERLY LOADED AND THAT THE CONFIGURATION MODULE IS FUNCTIONAL. SEE "SOFTWARE LOAD" ON PAGE 342501152, TO VERIFY SOFTWARE HAS BEEN PROPERLY LOADED. IN CONFIG MODE ON THE PFD USING THE FMS KNOB GO TO GDU GROUP, KEY TEST PAGE AND VERIFY THAT “IICEEPROM” BOX IS CHECKED GREEN TO VERIFY THE CONFIGURATION MODULE IS FUNCTIONAL.

NOTE: Fields / selections in (parentheses) within the unlock procedures below apply to S/N’s 4636633, 4636652 and up.

(a) CHARTVIEW Unlock (Optional)

1) Remove power from both PFDs and MFD by opening the PFD and MFD circuit breakers.
2) Insert the Chart Unlock card, in the upper slot of the PFD1.

NOTE: The Chartview Unlock card can only enable Charts on one system (one aircraft) and is locked to the system ID for that aircraft.

3) While holding the ENT key on both PFDs and MFD, restore power by closing the circuit breakers.
4) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the MFD and PFDs release the ENT key.
5) On the PFD1, go to the System Upload page.
6) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the AIRFRAME (GROUP) field. Press ENT.
7) Use the small FMS knob to create a pick list and select Enable Chartview in the FILE (ITEM) field. Press ENT.
8) Verify there is a check mark in the box in the configuration column for Airframe.
9) Press the LOAD softkey.
10) Monitor the status of the upload. When the upload is finished, Complete and Pass is displayed. Press ENT.
11) De-activate the cursor.
12) Power down the system and remove the Chartview Unlock card from the PFD.
13) After the Chart function is unlocked, the card should be stored with the aircraft records.

(b) TAWS Unlock (Optional)

1) Remove power from both PFDs and MFD by opening the PFD and MFD circuit breakers.
2) Insert the TAWS Unlock card in the upper slot of the PFD1.

NOTE: The TAWS Unlock card can only enable TAWS on one system (one aircraft) and is locked to the system ID for that aircraft.

3) While holding the ENT key on the both PFDs and MFD, restore power by closing the PFD and MFD circuit breakers.
4) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.
5) On the PFD1, go to the System Upload page.
6) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the AIRFRAME (GROUP) field. Press ENT.
7) Use the small FMS knob to create a pick list and select Enable TAWS in the FILE (ITEM) field. Press ENT.

8) Verify there is a check mark in the box in the configuration column for Airframe.
9) Press the LOAD softkey.
10) Monitor the status of the upload. When the upload is finished, Complete and Pass is displayed. Press ENT.
11) De-activate the cursor.
12) Power down the system and remove the TAWS Unlock card from the PFD.
13) After the TAWS function is unlocked, the card should be stored with the aircraft records.

(c) Synthetic Vision Unlock (Optional)

See "SVS/Pathways Activation" on page 342501168.

(d) Enhanced AFCS Unlock

Available in S/N's 4636633, 4636652 and up.

1) Remove power from the MFD and PFDs by opening the PFD and MFD circuit breakers.
2) Insert the Enhanced AFCS Enable card in the upper slot of the #1 PFD.

NOTE: The Enhanced AFCS Enable card can only enable TAWS on one system (one aircraft) and is locked to the system ID for that aircraft.

3) While holding the ENT key on both PFDs and MFD, restore power by closing the PFD and MFD circuit breakers.
4) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.
5) On the #1 PFD, go to the SYSTEM UPLOAD page with the small FMS knob.
6) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.
7) Use the small FMS knob to create a pick list and select Enhanced AFCS in the ITEM field. Press ENT.
8) Verify there is a check mark in the box in the configuration column for Airframe.
9) Press the LOAD softkey.
10) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.
11) De-activate the cursor.
12) Power down the system and remove the Enhanced AFCS Enable card from the PFD and store it with the aircraft records.

(8) Aviation Database Loading Procedures

(a) GARMIN GDU 1XXX Terrain Data Card Loading Procedure

NOTE: Garmin Field Service Engineering, resident at Piper, will be responsible for supplying updates for the GDU 1XXX Terrain Data Base Card.

1) Remove power from the PFDs and MFD by pulling the PFD and MFD circuit breakers.
2) Identify each Terrain Data Base Card respective to the display in which it is to be installed (PFD1, MFD, PFD2).
3) Install a Terrain Database Card in the bottom slot of the PFDs and MFD.
4) This completes the installation of the Terrain Database Card.

(b) JEPPS NAV and JEPPS CHARTS Database Loading Procedure

Loading the JEPPS NAV or JEPPS CHARTS database requires an appropriate data card containing the database. The data cards delivered with the airplane contain databases current at the time of delivery. Up-to-date databases are available from Jeppesen on a subscription basis.

1) JEPPS NAV Database Loading Procedure

a) Remove power from the PFDs and MFD by pulling the PFD and MFD circuit breakers.

b) Install the JEPPS NAV Database card in the top slot of PFD1.

c) Restore power to the appropriate display (PFD1, PFD2, MFD) by closing the appropriate circuit breaker.

d) S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153

1] The following prompt is displayed in the upper left corner of the display.

"DO YOU WANT TO UPDATE THE AVIATION DATABASE?"
"PRESS CLR FOR NO AND ENT FOR YES "
"YOU HAVE 30 SECONDS BEFORE NO IS RETURNED"

2] Press the ENT key to confirm the database update. The following prompt is displayed:

"DO YOU WANT TO UPDATE THE AVIATION DATABASE?"
"PRESS CLR FOR NO AND ENT FOR YES"
"YOU HAVE 30 SECONDS BEFORE NO IS RETURNED"

"UPDATING AVIATION DATABASE"

"UPDATED 2 FILES SUCCESSFULLY!"

e) S/N's 4636633, 4636652 and up

1] Press YES softkey after the following prompt is displayed on the #1 PFD:

DO YOU WANT TO UPDATE THE STANDBY NAVIGATION DATABASE?

THE STANDBY DATABASE WILL BE ACTIVATED UPON THE FIRST ON-GROUND POWER CYCLE ON OR AFTER 00:00 SYSTEM TIME ON THE EFFECTIVE DATE.

NO WILL BE ASSUMED IN 30 SECONDS.

2] Press the ENT key to confirm the database update. Press YES softkey after the following prompt is displayed:

DO YOU WANT TO UPDATE THE ACTIVE NAVIGATION DATABASE?

SELECTING YES WILL OVERWRITE THE ACTIVE NAVIGATION DATABASE.

NO WILL BE ASSUMED IN 30 SECONDS.

f) After the update completes, the PFD starts in normal mode. The PFD aviation (navigation) database is now updated.

g) Remove the JEPPS NAV Database card from the top slot of PFD1 and place it in the top slot of PFD2. Repeat step d) or e) for PFD2. After the update completes, PFD2 starts in normal mode. The PFD2 aviation (navigation) database is now updated.

h) Remove the JEPPS NAV Database card from the top slot of PFD2 and place it in the top slot of the MFD. Repeat step d) or e) for the MFD. After the update completes, the MFD starts in normal mode. The MFD aviation (navigation) database is now updated.

i) After the JEPPS NAV Database loading procedure is completed, the card should be stored with the aircraft records.

2) JEPPS CHARTS Loading Procedure

a) Remove power from the MFD by pulling the MFD circuit breaker.

b) Install Jepps Charts Card in the bottom slot of the MFD.

c) Restore power to the MFD by closing the MFD circuit breaker.

d) Press the ENT key to confirm the database update.

e) After the update completes, the MFD starts in normal mode.

f) The Jepps Charts are now loaded in the MFD and do not need to be loaded in the PFD.

g) The Jepps Charts Card remains in the bottom slot of the MFD.

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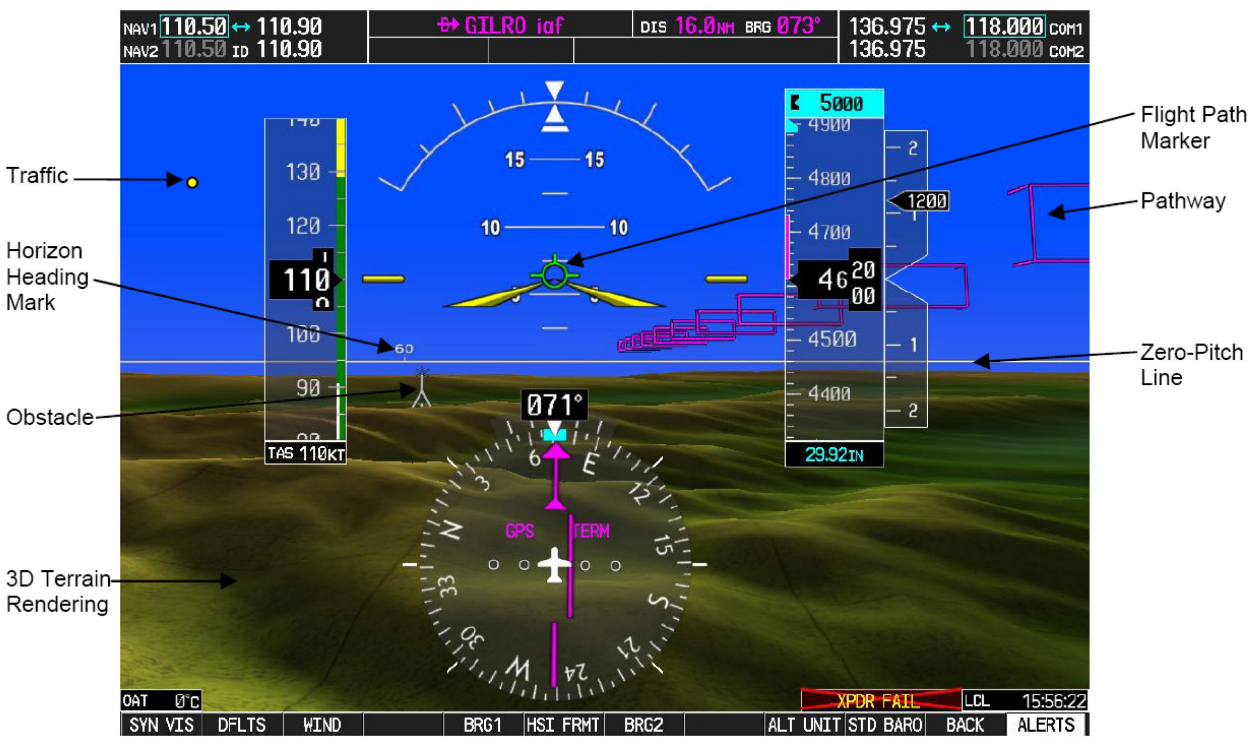
(9) Synthetic Vision System

(a) Description

The Synthetic Vision System (SVS) is intended to provide the crew with a greater awareness of the aircraft's position relative to surrounding terrain, obstacles, and traffic (optional). This is accomplished by placing a three dimensional depiction of terrain, obstacles, and traffic in the PFD primary field of view such that their proximity to the aircraft is more easily understood during instrument scanning. The display of SVS can be turned on and off by the pilot by a dedicated softkey on the PFD.

The SVS PFD frame-of-reference is aligned with the aircraft body frame, using AHRS attitude and heading data for orientation. Accordingly, the SVS is egocentric and attitude-aligned. GPS position and GPS-derived altitude are used for the SVS position and elevation references, respectively. The SVS 3D terrain presentation is generated from a high resolution 9 arc-second terrain database image. For the SVS system to be made available by the G1000, valid attitude, heading, GPS position, and terrain databases are required.

All previous existing PFD display features are retained and drawn over the SVS. For example, airspeed, altitude, vertical speed, heading, and HSI items will remain unchanged, and will be drawn over the SVS presentation.



SVS / Pathways Features
Figure 44

1) Synthetic Vision System

3D Terrain Rendering on PFD – SVS system will utilize an egocentric, forward looking perspective of 3D terrain rendering and shading, to be displayed on the PFD.

Zero Pitch Line – The PFD displays an artificial horizon pitch reference line above the display of terrain, similar to the existing artificial horizon which currently separates the blue/brown attitude depiction. The display of horizon heading marks in 30° intervals on the pitch line is a pilot-selectable feature.

Flight Path Marker – The projected aircraft flight path is displayed in the form of a Flight Path Marker on the PFD.

Display of Traffic – Traffic data from existing certified TIS, provided by the G1000 GTX 33 Mode-S transponder, is used to generate on-screen traffic symbols.

Display of Land-based Objects – Obstacles, runways, water bodies, and other land features are displayed on the PFD.

2) Pathways

SVS / Pathways are a pilot-selectable feature. The display of SVS can be turned on and off by the pilot by a dedicated softkey on the PFD. The display of Pathways requires that the 3D Terrain feature be displayed (i.e. SYN TERR softkey pressed).

When active, the Pathway is displayed on the PFD and will show flight path information in the form of a virtual 3D flight path, composed of rectangular elements which form a tunnel. The depiction of this pathway is derived from the existing FMS navigation capabilities of the G1000, and the display will be consistent with that of the active flight plan shown on the MFD.

3) Control & Operation

SVS/Pathways is an embedded software feature of the G1000 and is controlled by using PFD softkeys. Specific details on operating the SVS/Pathways feature can be found in the Synthetic Vision & Pathways AFMS specified in the Pilot's Operating Handbook (POH).

4) Servicing

The software upgrades and SVS/Pathways function are non-serviceable software which is either active or inactive in the G1000 system. Return to Service Procedure, below, provides a check to determine the state, and versions of software currently installed.

Reactivation may be required following normal maintenance procedures.

See "SVS/Pathways Activation" on page 342501168.

(b) Troubleshooting

The SVS/Pathways software feature requires the following G1000 sensors/data to be valid:

1) AHRS
2) Heading
3) GPS Position
4) 9 Arc-Second Terrain Data

In the event that one the above items fails or is unavailable, the SVS/Pathways feature is automatically removed from the PFD. "Chart 33" describes possible symptoms associated with the SVS/Pathways feature, and provides corresponding actions for troubleshooting.

"Chart 34" on page 342501167 provides SVS/Pathways specific alert messages which may appear in the Alerts Window on the PFD (press the ALERTS softkey on the PFD to view the Alerts Window).

CHART 33
TROUBLESHOOTING - SYNTHETIC VISION SYSTEM

Symptom	Recommended Action
"SYN VIS" softkey does not appear on PFD softkey tier.	Follow steps in SVS/Pathways Activation, below, to reactivate the SVS/Pathways feature.
3D terrain presentation does not appear on PFD.	Verify that P/N 010-00330-43 terrain datacards are installed in the lower slot of the PFD and MFD. Verify that no alert messages are shown in the PFD Alerts Window. Verify that the G1000 AHRS, and heading data are valid on the PFD. Verify that a valid GPS 3D position solution is being received. If a terrain database update has just been performed, allow the system time to initialize and verify the data. When the databases have been verified, the current database cycle and version are reported on the MFD AUX – System Status page.

CHART 34
SYNTHETIC VISION SYSTEM – ALERT MESSAGES

Failure Message	Cause	Solution
SVS – SVS DISABLED: Out of available terrain region.	SVS is disabled because the aircraft exceeded the boundaries of the loaded terrain database.	Geographical operation limitations are defined in the SVS/Pathways AFMS, specified in the POH.
SVS – SVS DISABLED: Terrain DB resolution too low.	SVS is disabled because a 9 Arc-Second or better database is not currently loaded.	Ensure P/N 010-00330-43 Terrain Cards are installed in the lower slot of each display. If terrain data has been recently updated, ensure that the correct 9 Arc-Second databases were used.

(c) Removal and Replacement

SVS/Pathways and all associated software version changes are non-serviceable software and cannot be physically removed or replaced in the G1000.

Reinstallation of the software upgrade and SVS reactivation should be accomplished as described below.

(d) Return to Service

1) SVS/Pathways Feature

Certain G1000 maintenance activities will require the SVS/Pathways feature to be reactivated and/or the unit software to be reloaded.

Specifically, any time the default AIRFRAME configuration file is reloaded to the G1000, the SVS/Pathways feature ‘activation’ is lost. See "Configuration File Descriptions" on page 342501146, for detailed information regarding G1000 configuration files.

The following specific maintenance scenarios will require re-activation of the SVS/Pathways feature:

a) Software and configuration files are completely re-loaded to the G1000 system.

b) Software and configuration files specific to the PFD(s) are re-loaded, either as the result of troubleshooting procedures or complete PFD replacement.

In the event one of these scenarios occurs, the technician must reactivate and verify the SVS/Pathways feature following the steps outlined below.

2) Software Reloading

Removal and replacement of any LRU mandates the reloading of the applicable LRU software. Once completed, the software and configuration must be loaded into the new unit following the steps outlined in the configuration and software loading procedures (i.e., "G1000 Software/Configuration Procedure" on page 342501150,"MFD & PFD Software Load" on page 342501152, "System Upload" on page 342501153, "Optional Equipment Configuration" on page 342501154, and "Software Load Confirmation" on page 342501156).

(e) SVS/Pathways Activation

This section provides the requirements and instructions necessary to unlock the G1000 Synthetic Vision / Pathways feature.

1) Database Cards

The Garmin Synthetic Vision and Pathways feature requires 9 arc-second high resolution terrain databases to function.

Each G1000 display must be equipped with the P/N 010-00330-43 Terrain/Obstacle/SafeTaxi database card installed in the lower slot.

2) SVS Activation Procedure

Activate the SVS/Pathways feature by performing the following steps:

NOTE: Fields / selections in (parentheses) below apply to S/N's 4636633, 4636652 and up.

a) Apply power to the G1000 system.
b) Pull the MFD, PFD1, and PFD2 circuit breakers.
c) A special SVS Unlock card is required to activate this feature, see above. Insert this card in the upper slot of PFD1 .

NOTE: The Synthetic Vision Unlock card can only enable Synthetic Vision on one system (one aircraft) and is locked to the system ID for that aircraft.

d) While holding the ENT key on the PFD1 , PFD2 and MFD (for MFD press and hold the farthest right pushbutton), restore power to the displays.
e) When the words **INITIALIZING SYSTEM** appear in the upper left corner of the displays, release the ENT key.
f) On PFD1, go to the System Upload page using the FMS knob.
g) Activate the cursor. Use the small FMS knob to select CONFIGURATION FILES in the AIRFRAME (GROUP) field and press the ENT key.
h) Highlight the FILE (ITEM) field. Use the small FMS knob to select the "Enable SVS Dual PFD" option and press the ENT key.
i) Verify there is a check mark in the box in the configuration column for Airframe.
j) Press the LOAD softkey.
k) When the upload is Complete and Pass is displayed, press the ENT key.
l) De-activate the cursor.
m) Power down the system and remove the SVS Unlock card from the PFD.
n) After the SVS is unlocked, the Synthetic Vision Unlock card should be stored with the aircraft records.

3) SVS / Pathways Installation Verification

a) Apply power to the G1000 system. Allow the AHRS and magnetometer systems to stabilize and align. Verify that air data information becomes valid on the PFDs. Check the MFD AUX – System Status page to verify GPS signals acquisition.
b) Press the ALERTS softkey on PFD1 and verify no database, manifest, or configuration errors exist.
c) Press the PFD softkey. Verify a SYN VIS softkey is shown in the lower left corner of the display.
d) Press the SYN VIS softkey, then press the SYN TERR softkey, to activate the Synthetic Vision terrain display feature. Verify that the traditional blue/brown attitude depiction is replaced with the Synthetic Vision rendering within 2–3 minutes of activation.
e) Installation is complete. Be sure to keep the SVS / Pathways unlock card with the aircraft for future use.

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M. Post-Installation Procedures

NOTE: This entire procedure must be successfully accomplished in order for the G1000 system to be airworthy in your airplane.

(1) Bus Power Check

NOTE: The following checks MUST be performed prior to electrical power-up with the associated wiring connected to the electronic equipment. Failure to perform these checks prior to powering up may result in damage to the equipment.

(a) Do not apply external power to the aircraft.

(b) Verify that all circuit breakers are set.

(c) Turn ON Battery Master switch with Avionics Bus Master switch and the Emergency switch in the “OFF” position.

1) Verify Avionics Bus 1, Avionics Bus 2, and Emergency Bus are not powered.
2) Pull #1 Main Bus circuit breaker and Non-Essential Bus circuit breaker.
3) Verify Aft Main Bus and Forward Main Bus are powered.
4) Verify Non-Essential Bus is not powered.
5) Reset #1 Main Bus circuit breaker. Pull #2 Main Bus circuit breaker.
6) Verify Aft Main Bus and Forward Main Bus are powered.
7) Reset Non-Essential Bus circuit breaker. Pull #1 Main Bus circuit breaker.
8) Verify Aft Main Bus, Forward Main Bus, and Non-Essential Bus are powered.
9) Reset #1 Main Bus circuit breaker and #2 Main Bus circuit breaker.

(d) Turn ON the Avionics Bus Master switch.

1) Pull #1 Avionics Bus circuit breaker, #1 Main Bus circuit breaker, #2 Main Bus circuit breaker, and Non-Essential Bus circuit breaker.
2) Verify Avionics Bus 1 and Avionics Bus 2 are powered.
3) Verify Aft Main Bus, Forward Main Bus, Non-Essential Bus, and Emergency Bus are not powered.
4) Pull the Avionics Bus Tie circuit breaker.
5) Verify Avionics Bus 2 is powered and Avionics Bus 1 is not powered.
6) Reset #1 Avionics Bus circuit breaker. Pull #2 Avionics Bus circuit breaker.
7) Verify Avionics Bus 2 is not powered and Avionics Bus 1 is powered.
8) Reset #2 Avionics Bus circuit breaker, #1 Main Bus circuit breaker, #2 Main Bus circuit breaker, Non-Essential Bus circuit breaker, and Avionics Bus Tie circuit breaker.

(e) Turn ON the Emergency switch then place Battery Master switch and Avionics Bus Master Switch in the “OFF” position.

1) Verify that the Main Busses (Forward and Aft) and Non-Essential Bus are not powered.
2) Verify that the Avionics Busses (#1 and #2) are not powered.
3) Verify that the Emergency Bus is powered.

(f) Turn OFF the Emergency switch, Battery Master switch and Avionics Bus Master Switch.

Verify that the Battery Bus is powered.

(2) Wiring Harness Checkout

The following is required only when a harness has been repaired, modified, or replaced, but is recommended when troubleshooting.

Prior to installing any LRUs, the wiring harness should be checked for proper connections to the aircraft systems and other avionics systems. Point to point continuity should be checked to expose any faults such as shorting to ground. Any faults or discrepancies found should be corrected before proceeding. Refer to wiring schematics in Chapter 91.

(3) Initial Display Testing

Verify the following:

(a) COM/NAV fields are valid in the top corners of the PFDs .

(b) Altitude, airspeed, vertical speed, heading and OAT fields are valid on the PFDs.

(c) No BACKUP PATHs alerts on the PFDs. If an LRU is not communication over its primary path, the BACKUP PATH alert will identify which LRU is having the problem.

(d) Select Reversionary Mode on both displays by pressing the display backup switches (located above each display), and displays will display valid attitude, altitude, airspeed, vertical speed and engine instruments. De-activate the Reversionary Mode.

(e) Verify flap position indication matches actual flap position for 0°, 10°, 20°, and 36°.

(f) Verify bus voltage indication.

1) With Battery Master switch ON and Emergency switch OFF.

2) Verify Main bus voltage indication.

3) With Battery Master switch OFF and Emergency switch ON.

4) Verify Emergency bus voltage indication.

(g) DOOR AJAR annunciation:

1) Open door.

2) Depress the door switches independently.

3) Verify “DOOR AJAR” annunciation does not clear from display.

4) Close door.

5) Verify “DOOR AJAR” annunciation is cleared from display.

6) Open baggage door.

7) Verify “DOOR AJAR” annunciation is displayed.

8) Close baggage door.

9) Verify “DOOR AJAR” annunciation is cleared from display.

(h) HYDR PUMP ON, GEAR DOWN, and VACUUM LOW (optional in PA-46R-350T) discrete I/O’s.

1) Place the airplane on jacks per 7-10-00.

2) Put PFD1 in configuration mode by pressing the ENT key while applying power.

3) Using the FMS knob on PFD1 go to the GIA I/O Configuration page (GIA group, page 3).

4) Activate the cursor and using the large FMS knob scroll to channel “IN 16”.

5) Cycle the landing gear and verify the DATA box next to CHANNEL “IN 16” is green during operation of the hydraulic pump and black when the hydraulic pump is off (this verifies the HYDR PUMP ON discrete I/O is functioning).

6) S/N’s 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

a) Using the large FMS knob on PFD1 scroll to channel “IN* 7A”.

b) Verify the DATA box next to CHANNEL “IN* 7A” is green with the landing gear down and black with the landing gear up (this verifies the GEAR DOWN discrete I/O is functioning).

c) Using the large FMS knob on PFD1 scroll to channel “IN* 3”.

d) Verify the DATA box next to CHANNEL “IN* 3” is green with low vacuum (this verifies the VACUUM LOW discrete I/O is functioning). (Optional in PA-46R-350T.)

7) S/N’s 4636633, 4636652 and up.

a) Using the large FMS knob on the #1 PFD scroll to channel “IN* 1A”. Use large FMS knob to scroll the list to check items below as needed.

b) Verify the DATA boxes next to CHANNEL “IN* 1A”, “10A”, “14A”, and “17A” are green with the landing gear down and black with the landing gear up (this verifies the GEAR DOWN discrete I/O is functioning).

c) Verify the DATA boxes next to CHANNEL “IN* 2A”, “11A”, “15A”, and “16A” are black with the landing gear down and green with the landing gear up (this verifies the GEAR DOWN discrete I/O is functioning).

d) Using the large FMS knob on the #1 PFD scroll to channel “IN* 3”.

e) Verify the DATA box next to CHANNEL “IN* 3” is green with low vacuum (this verifies the VACUUM LOW discrete I/O is functioning).

f) Scroll to the top of the page and change the UNIT to GIA2. Repeat steps c through j to verify GIA2 discretes.

g) Cycle power to #1 PFD. Power-up #1 PFD in normal mode.

(i) STALL WARN FAIL annunciation.

1) Pull the “STALL WARN” circuit breaker.

2) Verify “STALL WARN FAIL” annunciation appears on display.

3) Close “STALL WARN” circuit breaker.

4) Verify “STALL WARN FAIL” annunciation is cleared from display.

(j) PROP HEAT FAIL annunciation. (Optional)

1) Turn on Prop Heat and pull the “PROP HEAT” circuit breaker.

2) Verify “PROP HEAT FAIL” annunciation appears on display.

3) Close “PROP HEAT” circuit breaker.

4) Verify “PROP HEAT FAIL” annunciation is cleared from display.

(k) Pitot Heat annunciations.

1) S/N’s 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

a) Turn the Pitot Heat switch to the ON position.

b) Pull the “L PITOT HEAT” circuit breaker.

c) Verify “L PITOT HT FAIL” annunciation appears on display.

d) Pull the “R PITOT HEAT” circuit breaker.

e) Verify “R PITOT HT FAIL” annunciation appears on display

f) Close “L PITOT HEAT” and “R PITOT HEAT” circuit breakers.

g) Verify “R PITOT HT FAIL” and “L PITOT HT FAIL” is cleared from display.

2) S/N's 4636633, 4636652 and up.
   a) Turn the Pitot Heat switch to the OFF position.
   b) Verify "PITOT HEAT OFF" annunciation appears on display.
   c) Turn the Pitot Heat switch to the ON position.
   d) Verify "PITOT HEAT OFF" annunciation is cleared from display.
   e) Pull the "L PITOT HEAT" circuit breaker.
   f) Verify "L PITOT HT FAIL" annunciation appears on display.
   g) Pull the "R PITOT HEAT" circuit breaker.
   h) Verify "PITOT HEAT OFF" annunciation replaces "L PITOT HT FAIL" annunciation.
   i) Close "L PITOT HEAT" circuit breaker.
   j) Verify "R PITOT HT FAIL" annunciation replaces "PITOT HEAT OFF" annunciation.
   k) Close "R PITOT HEAT" circuit breaker.
   l) Verify "R PITOT HT FAIL" is cleared from display.
3) Turn OFF Prop Heat switch after test is complete.

(l) Avionics Cooling Fan Fail Annunciation Test
1) S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.
   a) Pull the "AV/MFD FAN" circuit breaker.
   b) Verify "AVIONICS FAN FAIL" and "MFD FAN FAIL" messages appear on PFD display.
   c) Close the "AV/MFD FAN" circuit breaker.
   d) Verify "AVIONICS FAN FAIL" and "MFD FAN FAIL" messages are cleared from the PFD display.
   e) Pull the "PFD2 FAN" circuit breaker.
   f) Verify "PFD2 FAN FAIL" message appears on the PFD display.
   g) Close the "PFD2 FAN" circuit breaker.
   h) Verify "PFD2 FAN FAIL" message is cleared from the PFD display.
   i) Pull the "PFD1 FAN" circuit breaker.
   j) Verify "PFD1 FAN FAIL" message appears on the PFD display.
   k) Close the "PFD1 FAN" circuit breaker.
   l) Verify "PFD1 FAN FAIL" message is cleared from the PFD display.
2) S/N's 4636633, 4636652 and up.
   a) Pull the "MFD FAN" circuit breaker.
   b) Verify "MFD FAN FAIL" advisory CAS on PFDs after 5 second delay.
   c) Close the "MFD FAN" circuit breaker.
   d) Verify "MFD FAN FAIL" advisory CAS is cleared from displays.
   e) Pull the "PFD2 FAN" circuit breaker.
   f) Verify "PFD2 FAN FAIL" advisory CAS on PFDs after 5 second delay.
   g) Close the "PFD2 FAN" circuit breaker.
   h) Verify "PFD2 FAN FAIL" advisory CAS is cleared from displays.
   i) Pull the "PFD1 FAN" circuit breaker.
   j) Verify "PFD1 FAN FAIL" advisory CAS on PFDs after 5 second delay.
   k) Close the "PFD1 FAN" circuit breaker.

l) Verify “PFD1 FAN FAIL” message is cleared from display.

m) Pull the “AV FANS” circuit breaker.

n) Verify “AV1 FAN FAIL” and “AV2 FAN FAIL” advisory CAS on PFDs after 5 second delay.

o) If installed, verify “AFT AV FAN FAIL” system message on PFDs.

p) Close the “AV FANS” circuit breaker.

q) Verify FAN FAIL advisories and system message are cleared.

(m) Starter Engaged Annunciation

S/N’s 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

1) With power removed from aircraft, disconnect P401 connector (just inside baggage compartment door, on upper edge of forward pressure bulkhead).

2) Apply ground power to aircraft.

3) Apply 28 volts to Pin T on J401.

4) After 30 seconds verify “STARTER ENGAGED” annunciation appears on display.

5) Remove ground power from aircraft and reconnect P401.

(n) STALL HEAT Annunciation

1) S/N’s 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

a) Turn on stall heat switch and verify “STALL HEAT ON” advisory CAS message.

b) Verify the stall vane is heated.

c) Turn off stall heat switch and verify “STALL HEAT ON” advisory CAS message clears.

2) S/N’s 4636633, 4636652 and up.

a) Turn on stall heat switch and verify “STALL HT INHIB” advisory CAS message.

b) Verify the stall vane is not heated.

c) Pull ADC1 and ADC2 circuit breakers. Verify “STALL HT INHIB” advisory is cleared.

d) Verify the stall vane is heated and “STALL HEAT FAIL” is not annunciated.

e) Reset ADC1 and ADC2 circuit breakers and turn off stall heat switch.

(4) Emergency Lighting Test

With engine off and no external power applied to aircraft select Emergency Switch “ON” then select Battery Master Switch and Avionics Switch “OFF”.

These items remain lit:

(a) S/N’s 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

1) Internal lights for the Standby ADI

2) Standby ADI On/Off switch light

3) Internal light for the whiskey compass

4) Internal lights for the Standby Airspeed

5) Internal lights for the Standby Altimeter

6) Master Caution Reset Switch Light
   a) Verify Master Caution Reset switch light illuminates.
   b) Push the Master Caution Reset Switch.
   c) Verify the Master Caution Reset switch light turns off.
7) Landing Indication Lights (R Main, L Main, and Nose)
8) PFD1
9) Audio Panel
10) All other flood lights and equipment lights extinguish.
11) All Overhead Switch Panel and Instrument Panel switch lights extinguish.

(b) S/N's 4636633, 4636652 and up.
1) EBD Standby Instrument
2) PFD1
3) Audio Panel
4) All other flood lights and equipment lights extinguish.
5) All Overhead Switch Panel and Instrument Panel switch lights extinguish.

(5) Emergency Bus Operation Verification
(a) Ensure Engine is “OFF”.
(b) Disconnect external power, if present.
(c) Turn Battery Master SWITCH “OFF“.
(d) Turn EMERGENCY switch “ON”.
(e) Select reversionary mode on PFD1 by pressing the display backup switch above the PFD.
(f) Verify all the following functions are operational:
1) PFD #1
2) COM #1
3) NAV #1
4) AHRS #1
5) Audio Panel
6) Air Data Computer #1
7) GEA 71 (Engine Instruments)
8) EBD Standby Instrument (S/N's 4636633, 4636652 and up.)

NOTE: MFD, PFD2, AHRS2, GDL 69A, Autopilot Controller, Transponder 1 and 2, Air Data Computer 2, Radar, Keypad, ADF, DME, TAS and Stormscope are not available. Also note that NAV2, GPS2, COM2, Oil Pressure, Vacuum (Mirage) / Surface Deice (optional), Differential Pressure (Mirage), and O2 (Matrix) display invalid.

(6) G1000 Checks
Perform the following steps to verify the following:

NOTE: Do not have External Power on the aircraft while conducting the following tests.

(a) AHRS and ADC data path check
1) In the GIA page group, go to the GIA RS-232/ARINC 429 CONFIG page.
2) With GIA1 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC1) and RS232 Channel 6 (GRS1) indicators are green.

3) With GIA1 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC1) and ARINC 429 IN6 (GRS1) indicators are green.

4) With GIA2 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC2) and RS232 Channel 6 (GRS2) indicators are green.

5) With GIA2 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC2) and ARINC 429 IN6 (GRS2) indicators are green.

6) In the PFD page group, go to the GIA RS-232/ARINC 429 CONFIG page.

7) With PFD1 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS1) and ARINC 429 IN2 (GDC1) indicators are green.

8) With PFD2 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS2) and ARINC 429 IN2 (GDC2) indicators are green.

(b) PFD and MFD Ethernet connection check

1) Re-start PFD2, PFD1 and MFD in normal mode.

2) On the right hand circuit breaker panel, open the MFD circuit breaker:

3) Verify NAV1 and COM1 remain valid and NAV2 and COM2 are each replaced by a red X on both PFD1 and PFD2.

4) Close the MFD circuit breaker and wait for MFD to initialize:

5) On the right hand circuit breaker panel, open the PFD2 circuit breaker:

6) Press the DISPLAY BACK UP button above the PFD1 display.

7) Verify NAV2, COM2, NAV1, and COM1 remain valid on both the MFD and PFD1.

8) Press the DISPLAY BACK UP button above the PFD1 display to return to normal mode.

9) Close the PFD2 circuit breaker and wait for PFD2 to initialize:

10) On the left aft circuit breaker panel, open the PFD1 circuit breaker:

11) Press the DISPLAY BACK UP button above the PFD2 display.

12) Verify NAV1 and COM1 are each replaced by a red X and NAV2 and COM2 remain valid on both the MFD and PFD2.

13) Press the DISPLAY BACK UP button above the PFD2 display to return to normal mode.

14) Close the PFD1 circuit breaker and wait for PFD1 to initialize.

(c) Perform the "GIA Failure Test" on page 342501214 to verify engine data availability to the displays and GPS data availability to the AHRS with either GIA inoperative.

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(7) G1000 Miscompare Checks

This procedure will check the AHRS, airspeed and altitude miscompare monitors. Accomplish the following checks with the aircraft positioned where it can receive GPS signals and magnetic heading.

Access to AHRS1 and AHRS2 will be required during this test.

(a) Ensure the G1000 is operating in normal mode.

(b) Connect a pitot/static test set to the aircraft ADC1 pitot and static ports (Pilot's side). (Do not connect the pitot/static tester to ADC2 ports at this time)

(c) Set the baro-correction on PFD1 and PFD2 to 29.92.

(d) On the AFCS mode controller, press the AP button and verify autopilot engages.

(e) Slowly increase the pitot/static test set to simulate an ADC1 / ADC2 altitude miscompare of greater than 200 ft., verify ADC2 altitude does not increase, autopilot does not disconnect and amber comparator window text "ALT MISCOMP" is displayed on PFD1 and PFD2.

(f) Reduce ADC1 altitude to ambient pressure.

(g) Repeat Step (e) with the pitot/static test set connected to the aircraft ADC2 pitot and static ports.

(h) Use the pitot/static test set to simulate an airspeed of 40 kts for ADC1 and 55 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(i) Use the pitot/static test set to simulate an airspeed of 55 kts for ADC1 and 40 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(j) Use the pitot/static test set to simulate an airspeed of 85 kts for ADC1 and 95 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(k) Use the pitot/static test set to simulate an airspeed of 95 kts for ADC1 and 85 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(l) Reduce ADC1 and ADC2 airspeeds to 0 kts.

(m) Slowly rotate AHRS1 along the lateral (pitch) axis to a pitch attitude of greater than five (5) degrees, verify the following:

1) AHRS2 pitch attitude does not change.

2) An amber "PIT MISCOMP" is annunciated on PFD1 and PFD2.

3) Autopilot disconnects at approximately 5 deg AHRS1 / AHRS2 miscompare.

4) The autopilot disconnect audio alert (two hi-low tones) sounds.

5) A red flashing AP annunciator on PFD1 and PFD2.

6) Flight director command bars remain in view with autopilot in HDG and PIT mode.

(n) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(o) Use the AFCS mode controller to re-engage autopilot.

(p) Slowly rotate AHRS1 along the longitudinal (roll) axis to a roll attitude of greater than five (5) degrees, verify the following:
   1) AHRS2 roll attitude does not change.
   2) An amber “ROLL MISCOMP” is annunciated on PFD1 and PFD2.
   3) Autopilot disconnects at approximately 5 deg AHRS1 / AHRS2 miscompare.
   4) The autopilot disconnect audio alert (two hi-low tones) sounds.
   5) A red flashing AP annunciator on PFD1 and PFD2.
   6) Flight director command bars remain in view with autopilot in HDG and PIT mode.

(q) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(r) Repeat Steps (m) through (q) for AHRS2.

N. LRU Test Procedures

(1) PFD/MFD Test

(a) Allow displays to initialize for ~1 minute.

(b) Check that all COM/NAV display fields are valid in the top corners of both PFDs.

(c) Check that attitude, heading, altitude, airspeed, vertical speed and OAT fields are valid within 2 minutes of power up on both PFDs.

(d) Press the SENSOR softkey on each PFD and switch between ADC1 and ADC2. Verify that data from both GDC 74As is valid on both displays.

(e) Press the SENSOR softkey on each PFD and switch between AHRS1 and AHRS2. Verify that data from both GRS 77s is valid on both displays.

(f) Check that the engine instrument fields are valid on the MFD.

(g) Push the red DISPLAY BACKUP button above the pilot-side PFD. Verify that the pilot-side PFD and MFD displays enter reversion mode. MFD should have valid altitude, airspeed, vertical speed, COM1, COM2, NAV1, NAV2 and engine instruments.

(h) De-activate pilot-side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD1 and MFD return to normal display modes.

(i) Repeat Step (a) using the co-pilot-side PFD. Ensure that PFD2 and MFD enter reversion mode and MFD displays valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(j) De-activate co-pilot's side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD2 and MFD return to normal display modes.

(k) If TAWS was activated, select the TAWS page (4th page in the MAP group) on the MFD.

1) Verify that the title at the top of the page reads "MAP – TAWS". If TAWS has not been enabled, the title will read "MAP – TERRAIN PROXIMITY" or "MAP – TERRAIN".

2) Press the MENU button and select "Test TAWS" from the pop-up menu.

3) After the TAWS test has completed, verify that "TAWS System Test Okay" is heard over the cockpit speaker.

(l) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(2) GMA 347 Test

S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

Except for marker beacon operation, an in-aircraft checkout may be performed in the aircraft with known good microphone, headset, and speaker.

(a) Check pilot and Copilot Intercom System (ICS) to verify the pilot and copilot MIC and phone jack operation.

(b) Select SPKR on the Audio Panel for two (2) seconds (for Passenger Address mode), press the MIC key and verify audio is heard over the cockpit speaker. (SPKR key flashes while in PA mode.)

(c) Verify operation of COM1. Conduct VSWR check per "Chart 35" on page 342501183.

(d) Failsafe Operation Check – GMA to COM #1

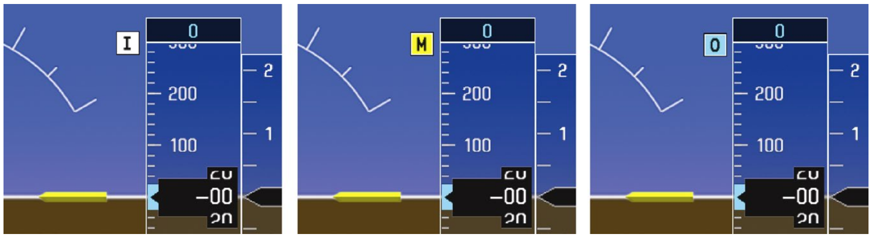
1) Turn the GMA OFF by pulling the AUDIO MKR circuit breaker.

2) Verify COM1 transmits and receives normally over pilot headset.

3) Verify COM1 audio is controlled by the PFD COM audio control.

4) Close the AUDIO MKR circuit breaker.

(e) Verify operation of COM2. Conduct VSWR check per "Chart 35" on page 342501183.



Marker Beacon Symbology
Figure 45

(f) Marker Beacon Test

1) Verify that the outer, middle, and inner markers appear on PFD1 and PFD2. See "Figure 45".

2) Verify Marker Beacon Audio and tones.

3) Verify the MKR MUTE key mutes the Marker Beacon Audio.

4) Verify Marker HI/LO Sensitivity operation.

(g) Verify Passenger ICS is working at all stations.

(h) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(3) GMA 350 Test

S/N's 4636633, 4636652 and up.

(a) External power applied to aircraft. Apply power to the avionics bus.

(b) Pilot and Co-Pilot ICS. (Verifies pilot and co-pilot mic and phone jack operation.)

(c) Pilot side only, hold PA key on the Audio Panel. On pilot yoke, press Mic Key and verify audio is heard over the cockpit speaker.

(d) Verify operation of COM 1.

Conduct VSWR check per "Chart 35" on page 342501183.

(e) Failsafe operation check for Audio Panel to #1 COM:

1) Pull AUDIO MKR circuit breaker.

2) Verify COM 1 transmits and receives normally over pilot headset.

3) Verify COM 1 audio is controlled by the PFD COM audio control.

4) Close the AUDIO MKR circuit breaker.

(f) Verify operation of COM 2. Conduct VSWR check per "Chart 35".

(g) Marker Beacon

Using a ramp tester, simulate the outer marker, middle marker and inner marker signals by following the test equipment manufacturer's instructions.

1) Verify outer, middle and inner markers appear on the PFD.

2) Verify Marker Beacon Audio and tones.

Verify that each marker audio signal is present over the pilot and co-pilot headphones and speaker.

3) Verify the MKR MUTE key mutes the Marker Beacon Audio.

(h) Verify Passenger ICS is working at all stations.

(i) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

CHART 35
VSWR TEST

VOLTAGE STANDING WAVE RATIO (VSWR)

The standing wave ratio is a measure of the amount of power transmitted to the antenna, compared to the amount reflected back to the transmitter. S.W.R. s are a function of the transmission line, the antenna and their installation.

A. Required Equipment

Thruline Wattmeter – Model 43 bird Electronics, Cleveland, Ohio.

B. S.W.R. Testing:

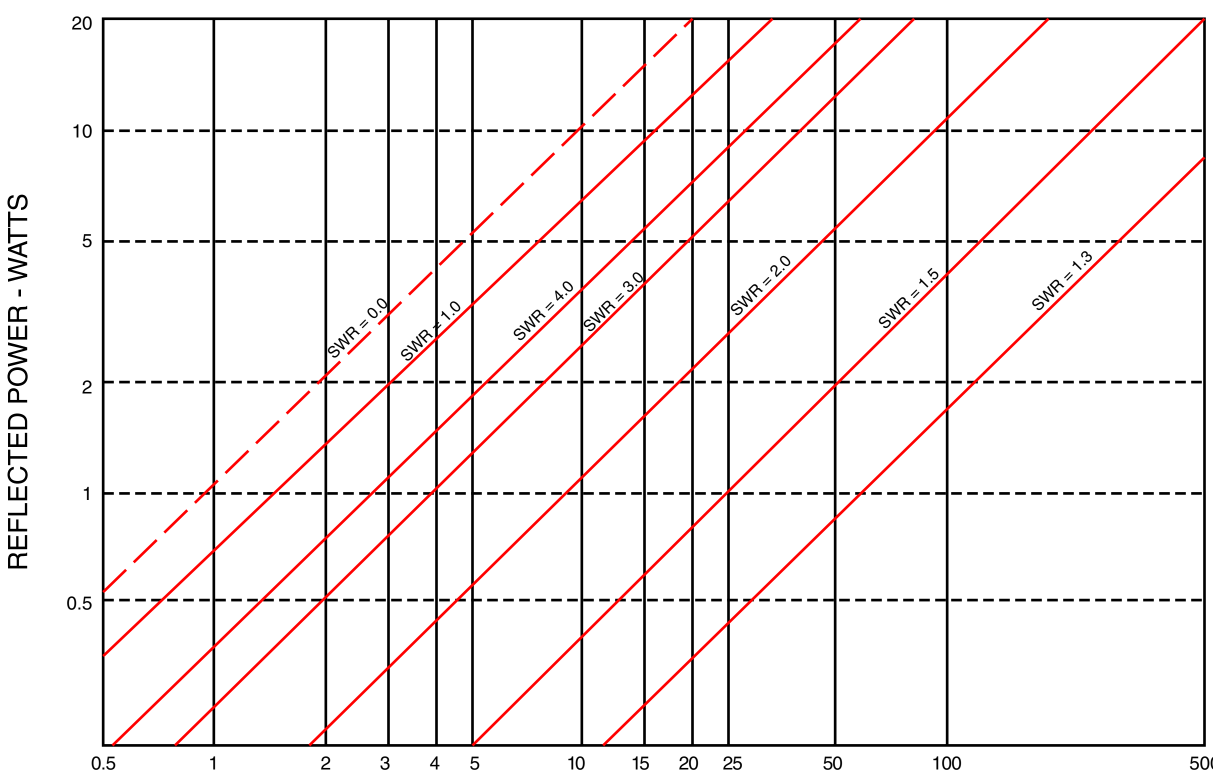
Test the antenna installation using the Thruline Wattmeter.

(1) Check VSWRs at 118.0, 123.0, 128.0, 132.0, and 135.95 MHZ by transmitting on COM 1 and then COM 2.

(2) Determine the VSWR value with the VSWR Conversion Chart below.

(3) VSWRs in excess of 2.5:1 are not acceptable.

VSWR CONVERSION CHART



(4) GIA 63W Test

(a) GPS Signal Acquisition

See "Figure 46".

The GIA 63W units should normally acquire a GPS navigation solution within 2 minutes of startup, provided the aircraft is outside (or indoors with a GPS repeater). Select the GPS STATUS page on the MFD (4th page in AUX group). Two softkeys on the bottom of the display allow the user to toggle between GPS1 and GPS2. Verify that both receivers show 3D DIFF NAV on the MFD.

Continue to the VHF COM Interference test.

NOTE: It may be necessary to temporarily disable or move away from GPS repeaters while testing, as repeaters may adversely affect GPS receiver performance.

(b) VHF COM Interference Test

This test must be conducted outside. Use of a GPS repeater inside a hangar may result in a failed test.

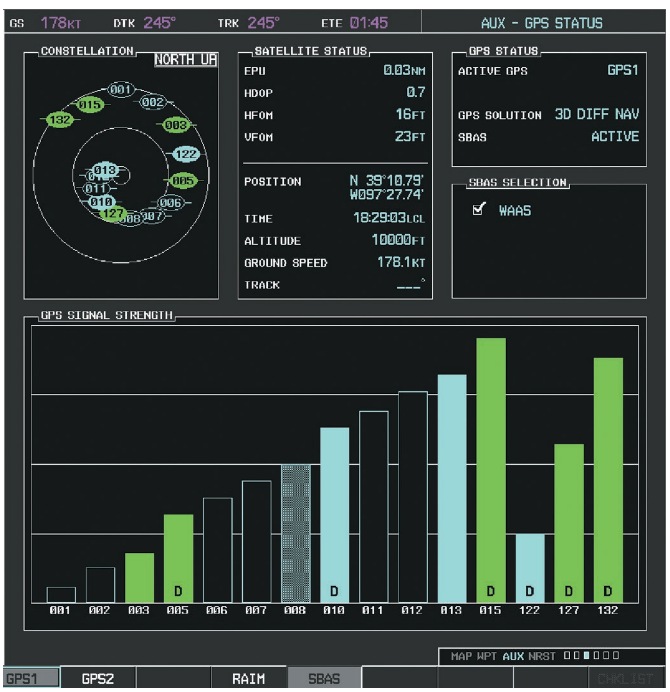
This procedure assumes that the system is currently set to 25 kHz COM channel spacing. Once the signal acquisition test has been completed successfully, perform the following steps:

1) On the MFD, monitor GPS signal strength bars on the AUX – GPS STATUS page.

2) On the PFD, ensure that the CDI is set to GPS. If it is not, press the ‘CDI’ softkey until GPS ENR is displayed.

3) Verify that the GPS “INTEG” flag is out of view.

4) Select 121.150 MHz on the COM1 transceiver.



GPS STATUS Page (MFD)
Figure 46

5) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
6) During the transmit period, verify that the GPS “INTEG” flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
7) Repeat steps 5 and 6 and re-transmit while monitoring GPS2 signal levels on the MFD.
8) Repeat steps 4 through 7 for each of the following frequencies:
   - 121.175 MHz
   - 121.200 MHz
   - 131.250 MHz
   - 131.275 MHz
   - 131.300 MHz
9) Repeat steps 4) through 8) for the COM2 transceiver (GIA2).
10) On the MFD, select the AUX – SYSTEM SETUP page.
11) Under the COM CONFIG field, change the COM channel spacing from 25 kHz to 8.33 kHz.
12) Go back to the AUX – GPS STATUS page.
13) Select 121.185 MHz on the COM1 transceiver.
14) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
15) During the transmit period, verify that the GPS “INTEG” flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
16) Repeat steps 14) and 15) and re-transmit while monitoring GPS2 signal levels on the MFD.
17) Repeat steps 14) through 16) for each of the following frequencies:
   - 121.190 MHz
   - 130.285 MHz
   - 131.290 Mhz
18) Repeat steps 14) through 17) for the COM2 transceiver (GIA2).
19) On the MFD, select the AUX – SYSTEM SETUP page and change the COM channel spacing back to 25 kHz.
20) Continue to the VOR/LOC/GS Test.

(c) VOR/LOC/GS Test

NOTE: The PFD HSI does not show a course deviation bar unless a valid VHF NAV frequency is tuned.

Check VOR, ILS, and Glideslope functions with ramp test equipment. Operate the equipment according to the test equipment manufacturer’s instructions. Adjust the RF signal to a level adequate to perform the test. Select the appropriate HSI source by using the CDI softkey.

1) Perform NAV 1 and NAV 2 accuracy tests.
   a) Full scale.
   b) Half scale.
   c) OBS operation.
2) Check NAV 1 and NAV 2 audio and Ident.
3) Check NAV 1 and NAV 2 Sensitivity.
4) Perform LOC 1 and LOC 2 accuracy tests.
   a) Full scale.
   b) Half scale.

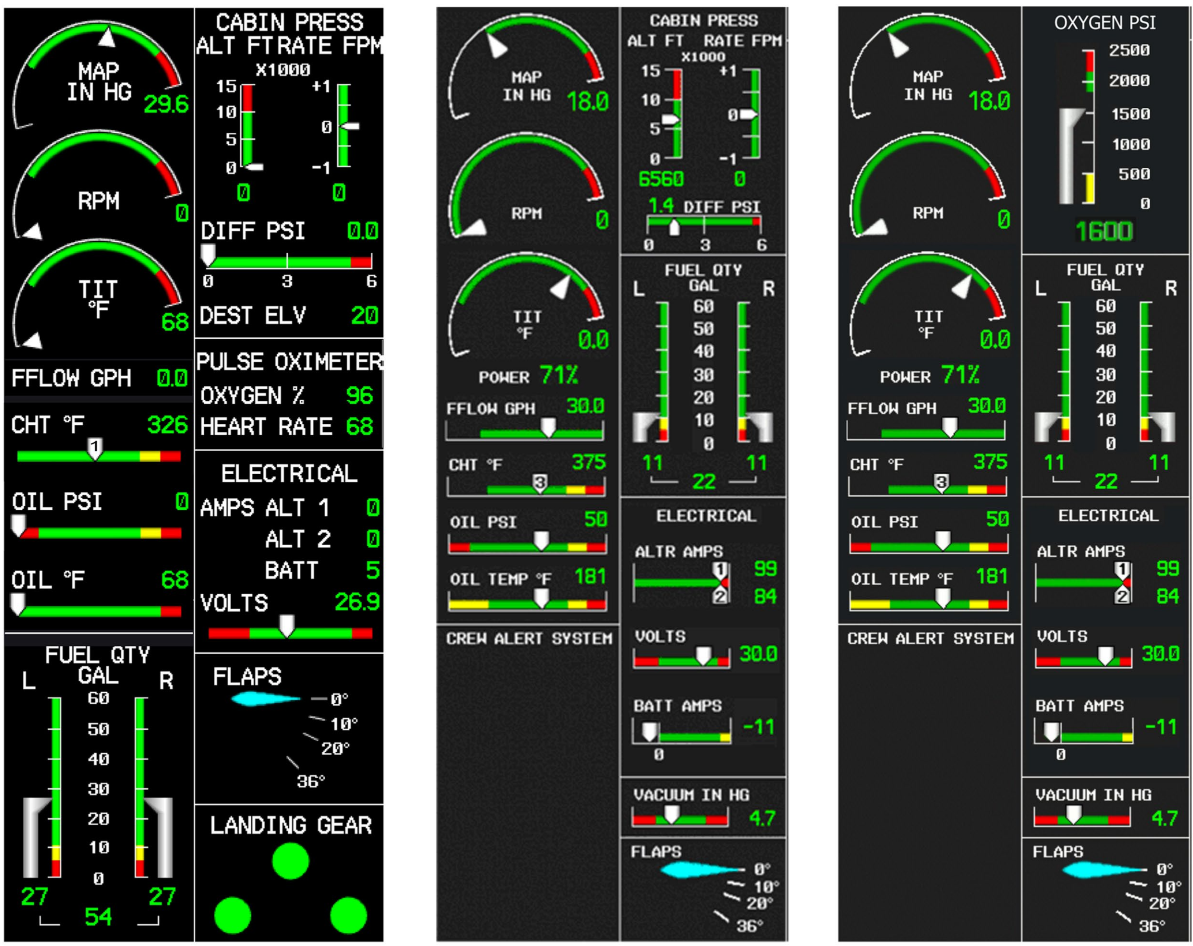
5) Check LOC 1 and LOC 2 Sensitivity.
6) Perform GS 1 and GS 2 accuracy tests.
   a) Full scale.
   b) Half scale.
7) Check GS 1 and GS 2 Sensitivity.

(5) GFC 700 Test
See Ground Checks under Garmin GFC 700 in 22-10-00.

(6) GEA 71 Test

(a) On the MFD (normal mode), check the indication for each of the sensor or monitor inputs with the aircraft engines off. In general, verify all engine and system instruments show valid static normal values and markings, with no red Xs or erratic indications. See "Figure 47" (baseline configuration instruments shown) for normal engine instrument markings.

(b) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.



S/N's 4636633, 4636652 and up

S/N's 4636460, 4636463–4636651, less 4636481, 4636633

PA-46-350P **PA-46R-350T**

Normal Engine Instruments
Figure 47

(7) GDC 74A Air Data Computer Tests

NOTE: Allow the unit to warm up for 15 minutes before performing the following tests.

Verification of the altimeter and airspeed must be performed using an air data test set (ADTS). The static port and altimeter must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) § 91.411 and Part 43 Appendix E. The PFD must be in Configuration mode and the MFD must be in Reversionary mode for performing the tests as outlined in Part 43 Appendix E.

(a) Part 43 Appendix E Testing

1) Preparation

CAUTION: CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

a) Start the G1000 system in normal mode.

b) Remove power to PFD1.

c) Turn PFD1 on in Configuration mode by pressing and holding the ENT key on the PFD while applying power.

d) Release the ENT key after “INITIALIZING SYSTEM” appears in the upper left corner of the PFD.

NOTE: Configuration mode contains certain pages and settings that are critical to aircraft operation and safety. These pages are protected and cannot be modified, unless the technician is properly authorized and equipped. However, most protected pages are viewable to allow system awareness for troubleshooting.

2) Testing

a) Using the FMS knob on the PFD turn to the GRS page group. Use the B ALT field for all CFR Part 43 Appendix E tests for G1000 altitude.

b) Place the MFD in Reversionary mode by pressing the DISPLAY BACKUP button above the respective PFD. The baro setting can then be read from the MFD for CFR Part 43 Appendix E tests.

NOTE: The baro setting on the MFD is controlled by PFD1. The baro setting will apply to both GDC1 and GDC2 regardless of the GDC selected on the MFD. The copilot’s display can be ignored.

(b) After completing the tests specified by § 91.411 and Part 43 Appendix E, return both the MFD and the PFD to normal mode.

NOTE: The following tests are above and beyond the requirements set forth in Appendix E, and are required only when appendix E tests are required.

(c) Pitot/Static Airspeed Test

1) Command air data test set (ADTS) to simulate air speeds shown in the table below.

Calibrated Air Speed (Knots)	Allowed tolerance (± Knots)
50	5.0
80	3.5
100	2.0
120	2.0
150	2.0

2) Wait for ADTS to report that target values have been achieved.

3) Verify that computed air speeds shown on the PFD are within the tolerances specified in the OEM maintenance documentation.

(d) Static Port Vertical Speed (Rate of Climb) Test

1) Command ADTS to change the altitude at the rates shown in the table below.

2) Wait for ADTS to report that target rates have been achieved.

3) Verify that the Rate of Climb reported by the Vertical Speed field on the PFD is within the tolerances specified in the table below:

Vertical Speed (feet/minute)	Allowed tolerance, (± feet/minute)
2000	100
0	45
-2000	100

(8) OAT Probe Check

Ensure on-side sensors for PFD1 and PFD2. Ensure the outside air temperature (OAT) probes and a calibrated thermometer stabilize at ambient temperature. Verify that the OAT measurement shown on PFD1 and PFD2, in degrees Celsius, indicate within 2°C of the ambient temperature as measured by the calibrated thermometer.

If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(9) GRS 77 / GMU 44 Calibration Procedures

See "Chart 36" to determine when various calibrations are required. The following calibration procedures are provided for the GRS 77 AHRS and GMU 44 Magnetometer:

- "GRS 77 Pitch/Roll Offset Calibration (Procedure A)"
- "GRS 77/GMU 44 Magnetometer Calibration (Procedure B)" on page 342501193
- "Engine Run-Up AHRS Vibration Test (Procedure C)" on page 342501195
- "Magnetometer Interference Test" on page 342501196

When ready to perform the procedures, shut the PFDs and MFD off by pulling the PFD and MFD circuit breakers. Restart all displays in configuration mode. Follow the steps given for each procedure on-screen at the GRS/GMU CALIBRATION page. Note that the CALIBRATE command cannot be selected and activated until the installer acknowledges all required steps have been carried out by pressing the ENT key on each step.

CHART 36
G1000 - GRS 77 AHRS/GMU 44 MAGNETOMETER CALIBRATIONS

Required GRS/GMU Calibrations			
Condition	**Calibrations Required**		
	Procedure A: GRS 77 Pitch/Roll Offset	Procedure B: GRS/GMU Magnetic Calibration	Procedure C: Engine Run-up Vibration Test
Either GMU 44 was removed and reinstalled. (no change in serial number)	None Required. Continue to GRS/GMU Test section.		
GMU 44 was replaced with new unit. (New serial number)		X	
GRS 77 AHRS was removed and/or replaced. The mounting tray was NOT removed and the mounting tray bolts were NOT loosened.	None Required. Continue to GRS/GMU Test section.		
GRS 77 AHRS was removed and/or replaced. The mounting tray WAS removed and/or mounting tray bolts WERE loosened.	X	X	X
GRS 77 AHRS Configuration Module was replaced.	X	X	X

(10) GRS 77 Pitch/Roll Offset Calibration (Procedure A)

This procedure must be performed for both GRS 77 units installed in the aircraft. This first procedure must be carried out with the engine OFF.

(a) Level the aircraft to within ±0.25° of zero pitch and zero roll using a digital level. (Refer to 8-20-00.)

(b) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(c) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(d) Select the GRS/GMU Calibration page by rotating the small FMS knob.

(e) Press the following softkeys in sequence on each PFD access the Pitch/Roll Calibration:

- 9
- 10
- 11
- 12 (far right softkey)

(f) Initiate the AHRS Ground Pitch/Roll Aircraft Level compensation mode by performing the following steps:

1) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 77 #1 ON PFD1 AND GRS 77 #2 ON PFD2.

2) Highlight the SELECT PROCEDURE window and select **PITCH/ROLL OFFSET** on each PFD.

3) Press the ENT key.

4) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

5) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

6) After several seconds, a new checklist appears in the lower half of the PFD. Press the ENT key as each step is confirmed. When the CONFIRM AIRCRAFT IS LEVEL field is blinking on both PFDs, press the ENT key to continue.

(g) The result of the pitch/roll offset compensation is displayed on each PFD. If successful, the PFDs report the successful calibration and return to normal operation.

(h) Press the ENT key on each PFD to conclude this procedure.

(i) Restart all displays in normal mode.

NOTE: The Magnetometer Calibration procedure that follows, must be carried out at a site that is determined to be free of magnetic disturbances. If it is unsure whether the site is 'clean', the technician should verify that the site is 'clean' by following the guidance provided. The technician may skip the Compass Rose Evaluation Procedure if the site condition is acceptable.

(11) Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure (Optional)

NOTE: Typically, a compass rose is an acceptable location to perform the magnetometer calibration procedure. However, because not all compass roses are well maintained, even an existing compass rose should be regularly evaluated using the method described here to determine if it is free of magnetic disturbances. If evaluation of an existing compass rose indicates that magnetic disturbances are present, then an alternative location must be found to perform the Magnetometer Calibration procedure.

A G1000-equipped airplane that has completed the pitch/roll offset compensation procedure can be used to evaluate a candidate site for magnetic disturbances and determine whether it is a suitable location to perform the magnetometer calibration procedure. The magnetometer calibration procedure itself contains the logic to simultaneously survey the location for magnetic cleanliness while it is computing the magnetometer calibration parameters. In order to evaluate a candidate site, the Magnetometer Calibration procedure must be performed twice: once turning clockwise around the site, and once turning counter-clockwise. Both times, the procedure should be conducted as described in "GRS 77/GMU 44 Magnetometer Calibration (Procedure B)", below, with the exception of the direction of turns around the site.

NOTE: Although GRS 77/GMU 44 Magnetometer Calibration (Procedure B) indicates that the Magnetometer Calibration procedure should be performed by making a series of clockwise turns around the site, the procedure can also be performed by making counterclockwise turns for the purpose of evaluating the site for magnetic disturbances.

If, upon completion of the Magnetometer Calibration procedure in both clockwise and counter-clockwise directions, the PFD displays the "CALIBRATION SUCCESSFUL / SITE IS CLEAN" message, then the candidate site is sufficiently free of magnetic disturbances and is acceptable for performing the Magnetometer Calibration procedure. It is important to obtain successful result in both the clockwise and counter-clockwise directions to ensure that the magnetometer sweeps over a large enough area at the candidate site.

If, upon completion of the Magnetometer Calibration procedure in either of the two directions, the PFD displays either the "MAG FIELD AT SITE NOT UNIFORM", or "MAG FIELD AT SITE DIFFERS FROM IGRF MODEL" message, then the site contains magnetic disturbances that are too large.

(12) GRS 77/GMU 44 Magnetometer Calibration (Procedure B)

NOTE: Procedure A must first be successfully accomplished before performing Procedure B, only for situations where the GRS 77 was replaced with a new unit.

Calibration Procedure B must be carried out on a compass rose in order to guarantee measurements free of environmental magnetic disturbances. Attempting to carry out this maneuver on a typical ramp area may not yield a successful calibration. The accuracy of the AHRS cannot be guaranteed if this calibration is not performed on a magnetically clean compass rose. If the compass rose condition is not known, it is recommended that the technician follow the guidance in "Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure" on page 342501192, above.

(a) Start the aircraft engine following the procedures in the appropriate POH.

(b) After aircraft engine startup, taxi the aircraft to a properly calibrated compass rose.

(c) At the compass rose, align the aircraft to a heading of magnetic north (±5°).

CAUTION: CALIBRATION PROCEDURE B MUST BE CARRIED OUT ON A COMPASS ROSE IN ORDER TO GUARANTEE MEASUREMENTS FREE OF ENVIRONMENTAL MAGNETIC DISTURBANCES. ATTEMPTING TO CARRY OUT THIS MANEUVER ON A TYPICAL RAMP AREA MAY NOT YIELD A SUCCESSFUL CALIBRATION. THE ACCURACY OF THE AHRS CANNOT BE GUARANTEED IF THIS CALIBRATION IS NOT PERFORMED ON A MAGNETICALLY CLEAN COMPASS ROSE OR EQUIVALENT.

NOTE: This procedure provides instructions for calibrating both GRS 77 magnetometers simultaneously by putting both PFD1 and PFD2 in configuration mode and following the procedure below, using PFD1 to calibrate GRS 77 #1 and PFD2 to calibrate GRS 77 #2.

(d) Start (restart) the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(e) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(f) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

- 9
- 10
- 11
- 12 (far right softkey)

(g) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select either GRS 77 #1 on PFD1 or GRS 77 #2 on PFD2, or both.

(h) Press the ENT key.

(i) Highlight the SELECT PROCEDURE window and select MAGNETOMETER on each PFD.

(j) Press the ENT key.

(k) Use the cursor to highlight the BEFORE CALIBRATION window on each PFD.

(l) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

(m) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(n) The PFD displays advise the operator when to turn the aircraft, when to stop, and when to turn again.

(o) Upon instruction to turn, taxi the aircraft in a right turn. After approximately 25° to 30° of turn from the last heading, the PFD displays advise the operator to stop the aircraft.

NOTE: Due to the difficulties in executing smooth, accurate turns, the PFDs may incorrectly interpret a station and instruct to “HOLD POSITION” prior to full completion of a 30° turn. If this scenario is encountered, it is best for the operator to ignore the “HOLD POSITION” command and instead use outside references to complete the approximate 30° turn. Instead of using the PFD instructions to turn as a real-time indication of when to turn, simply judge the 30° (±5°) turn increments of the aircraft by using the compass rose radials. Dwelling at these 30° increments for the time recommended by the PFDs should result in successful calibration.

(p) The PFDs guide the operator to dwell at multiple headings around a complete circle.

NOTE: Due to high winds or excessive airframe vibration, the operator may encounter a condition where a PFD restarts the 18-second countdown without full completion of the previous countdown. If this is encountered more than once for a given station, the operator should begin turning to the next station (approximately 30°). A minimum of 2 successful stations per quadrant (and for each GRS 77) is required, where a successful station is a full 18-second countdown followed by instruction to move. Ensure that at least 2 stations per quadrant are completed. Thus, it may sometimes be required to dwell at a station after a countdown restart. A maximum of 20 stations is allowed for the entire calibration procedure. If too many countdown restarts are encountered, the calibration will fail with the message, “TOO MANY STATIONS.”

(q) Repeat the turn-and-stop process until each PFD advises that a successful calibration is complete. The GRS 77 AHRS UNITS then enter the normal operational mode. Press the ENT key on each PFD to conclude this procedure.

(r) Repeat steps (d) through (q) for GRS2, if not performed concurrently.

(13) Engine Run-Up AHRS Vibration Test (Procedure C)

NOTE: Calibration Procedure C is performed in order to guarantee that the AHRS mounting is sufficiently rigid and insensitive to vibration. This procedure must be performed for both GRS 77 units installed in the aircraft. Calibration Procedures A and B are not required prior to this procedure.

(a) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(b) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(c) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

- 9
- 10
- 11
- 12 (far right softkey)

(d) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 77 #1 on PFD1 and GRS 77 #2 on PFD2.

(e) Highlight the SELECT PROCEDURE window and select ENGINE RUN-UP TEST on each PFD.

(f) Press the ENT key again on each PFD to select the first checklist item displayed.

(g) Follow the checklist items displayed on each PFD, and press the ENT key as each one is completed or confirmed. When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(h) The PFD displays instruct the operator to gradually increase power from idle to full throttle and back to idle over a period of 1–2 minutes.

(i) When the engine run-up is completed and the engine is back to an idle setting, press the ENT key on each PFD to indicate that the process is complete. When this is done, the TEST COMPLETE field stops blinking.

(j) Each PFD will state if the installation has passed or failed the vibration test. If the test fails, the specific measurements causing the failure are identified and associated numeric values are displayed on the PFD corresponding to each GRS 77 unit.

NOTE: Should a failure occur, the technician may perform the Engine Run-up test up to 3 times successively before corrective action must be taken. If the test does not pass after three attempts, then the installation should not be considered reliable until the source of the vibration problem is identified and remedied. In the event of repeated failure of the engine run-up test, record the values that are reported to be out of range for future reference.

The following are potential causes for failure of the engine run-up test:

1) Vibration motion of GRS 77 and/or GMU 44 caused by neighboring equipment and/or supports.
2) Mounting screws and other hardware for GRS 77 and/or GMU 44 not firmly attached.
3) GRS77 connector not firmly attached to unit.
4) Cabling leading to GRS 77 or GMU 44 not firmly secured to supporting structure.
5) An engine / propeller that is significantly out of balance.

(k) Press the ENT key on each PFD to conclude this procedure.

(14) Magnetometer Interference Test

A magnetometer interference test is available for troubleshooting and/or verifying a magnetically ‘clean’ installation of the GMU 44. This test exercises various devices on the aircraft that could potentially affect the magnetic field as measured by the GMU 44 (examples include navigation lights, control servos, landing gear motors, etc).

NOTE: This test is used to validate that no electronic device interferes with the operation of the GMU 44 magnetometer. It is highly recommended that this test be performed after installation or maintenance of electrical components on the aircraft and/or for troubleshooting the GMU 44.

NOTE: External power cart and other aircraft need to be as far away from the aircraft as possible.

NOTE: Items in [brackets] below apply to S/N’s 4636663, 4636652 and up.

(a) Turn Battery Master ON, AV BUS Master ON, Autopilot ON (AP engaged to allow pitch servo movement), and all circuit breakers in. Verify the Air Conditioning Condenser Blower Motor is OFF. [Verify the rudder and rudder trim tab are in the streamlined position.]

(b) Start the PFDs and MFD in the Configuration Mode by holding ENT on each display while closing the circuit breaker

(c) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(d) Select the GRS/GMU Calibration page, as shown in "Figure 48" on page 342501197, by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

- 9
- 10
- 11
- 12 (far right softkey)

(e) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 77 #1 on PFD1 and GRS 77 #2 on PFD2.

(f) Highlight the SELECT PROCEDURE window and select MAGNETOMETER INTERFERENCE TEST on each PFD.

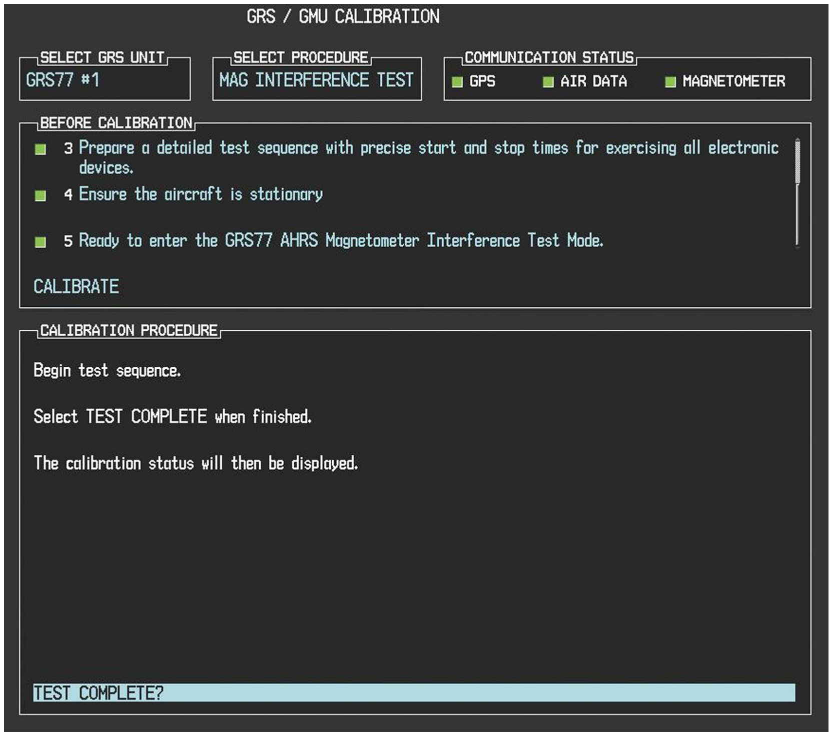
(g) Press the ENT key to select the first checklist item displayed on each PFD. Press the ENT key as each one is completed or confirmed (turns green). Have a stopwatch ready to begin recording the elapsed time.

NOTE: The list of relevant electronic devices are given in "Chart 37" on page 342501197. Use a stopwatch to accurately time the tests. All actions are to be carried out in the order and at the precise elapsed time as specified in the prepared test sequence. [MFD must be out of configuration mode prior to test being performed.]

(h) When the CALIBRATE field is blinking on both PFDs, press the ENT key to begin the procedure.

NOTE: It is important that the “time equals zero” moment corresponds with the moment the PFD first displays the blinking TEST COMPLETE? message.

(i) When the test is complete, press the ENT key on each PFD to indicate the process is complete. When this is done, the TEST COMPLETE annunciation stops blinking.



Magnetometer Interference Test
Figure 48

CHART 37
G1000 - MAGNETOMETER INTERFERENCE TEST

MAGNETOMETER INTERFERENCE TEST	
ELAPSED TIME Since start of test (Minutes:Seconds)	**ACTION**
0:00	Test Begins
0:10	Turn on strobe light
0:20	Turn off strobe light
0:30	Turn on fin strobe
0:40	Turn off fin strobe
0:50	Turn on nav lights (if optional tail light installed)
1:00	Turn off nav lights (if optional tail light installed)
1:10	Blower Motor on Low
1:20	Blower Motor off
1:30	Blower Motor on High
1:40	Blower Motor off

(j) Each PFD will display if the corresponding GRS 77 installation has passed or failed the magnetometer interference test. If the installation does not pass contact Piper Customer Support.

1) If the test fails, the installation should be considered unreliable until the source of magnetic interference is identified and remedied. When the magnetometer interference test fails, record the three magnetometer maximum deviation values and their corresponding timestamps. Any maximum deviation value greater than 2.5 milliGauss indicates a problem that must be resolved. Compare the corresponding timestamps with the prepared test sequence to identify which action produced the problem.

2) Two common reasons for a failed magnetometer interference test are: 1) new equipment is installed in close proximity to the GMU 44 magnetometer, and 2) an existing or new electronic device has become grounded through the aircraft structure instead of via the proper ground wire in a twisted shielded pair.

(k) Press the ENT key on each PFD to conclude the test.

(15) GRS/GMU Test

The aircraft can now be taxied back and the engine can be shut down for final testing. Restart the displays in normal mode to conduct final system checks. When the PFDs power up in normal mode, the AHRS attitude and heading information displayed should become valid within 1 minute of power-up, as shown in "Figure 49" (provided both GPS receivers have a valid position; if GPS is unavailable, AHRS initialization may take as long as 2 minutes).

If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

Normal Mode AHRS Check
Figure 49

(16) GDL 69eA Test

If installed and the XM Satellite Radio subscription is activated and current, proceed as follows:

(a) Power up the G1000.

(b) On the MFD, select “AUX - XM RADIO”.

(c) Increment and Decrement the XM channels using the MFD controls.

(d) Increase and Decrease the volume using the MFD controls.

(e) If GRC 10 installed:

1) Increment and Decrement the XM channels using the GRC10.

2) Increase and Decrease the volume using the GRC10.

(f) When (c) and (d), and if required (e), are complete.

1) Select an active music channel when complete.

2) Set the volume in the mid position when complete.

(g) Verify XM audio operation at pilot, copilot and all passenger stations. Verify stereo operation.

(h) Plug an external source into the AUX audio jack in the cabin and verify that it can be heard in all passenger stations. Verify that it is not audible via the pilot and copilot phone jacks.

CAUTION: STALL WARNING VANE IS HEATED. CARE MUST BE TAKEN TO AVOID INJURY.

(i) While monitoring XM audio through the crew headsets, activate the stall warning by using a screwdriver (or similar tool) to gently lift the heated stall warning vane on the leading edge of the wing (Use caution to not bend the vane). Verify the XM audio is muted while the stall warning is active.

(j) Landing Gear Aural Alert

In S/N’s 4636460, 4636463–4636651, less 4636481 and 4636633; 4692134 and up, less 4692141, 4692149, 4692153.

While monitoring XM audio through the crew headsets, activate the Gear Warning by activating the LTS/GEAR ANNUN TEST switch above the pilot-side PFD. Verify the XM audio is muted while the landing gear aural alert is active.

(k) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(17) GCU 476 Test

Perform the following key and knob presses and knob rotations on the GCU 476, and verify the actions on the MFD.

(a) Rotate the large FMS knob and verify that the page groups change.

(b) Rotate the small FMS knob and verify that the pages change within the page groups.

(c) Use the large FMS knob to display the MAP page group and the small FMS knob to display the NAVIGATION MAP page.

(d) Rotate the RANGE knob to the right and verify the map display zooms out.

(e) Rotate the RANGE knob to the left and verify the map display zooms in.

(f) Press the RANGE knob to get the pointer on the map display.

(g) Move the RANGE knob to the left and verify the pointer moves to the left.

(h) Move the pointer up, right and down and verify that the pointer moves accordingly.

(i) Press the RANGE knob to stop displaying the pointer.

(j) Press the left and right arrowheads of the SOFTKEY SELECT keys. Verify softkeys highlighting on the MFD changes.

(k) Press the SEL key to select one of the softkeys.

(l) Press the key to display the DIRECT TO page.

(m) Use the keypad to type KIXD and verify KIXD is displayed on the MFD.

(n) Press the SPC key to add a space and then the BACK key to delete the space.

(o) Press the CLR key to clear the field.

(p) Type K34 and then press the ENTER key twice. Verify that the flight path to K34 is displayed on the map.

(q) Press FPL key to open the ACTIVE FLIGHT PLAN page. Press FPL key again to close it.

(r) Press PROC key to open the PROCEDURES page. Press PROC key again to close it.

(s) Press the MENU key to open the MENU page. Press the MENU key again to close it.

(t) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(18) GMC 710 Test

Perform a basic operational check on the GMC 710. The following knob, wheel and key presses and rotations are to be performed on the GMC 710.

(a) Ensure the G1000 is operating in normal mode and the autopilot is operational.

(b) Press the FD key a few times; verify the Flight Director display on PFD1 toggles on and off. Leave the flight director displayed.

(c) Press the XFR key and verify the white-illuminated arrowhead points in the opposite direction.

(d) Verify the green arrow displayed at the top of PFD1 also points in the same direction.

(e) Rotate the ALT SEL knob and verify the altitude bug, displayed on PFD1 altitude tape, moves.

(f) Rotate the UP/DN wheel and verify the flight director moves in the vertical direction.

(g) Press the HDG key and verify the white illumination appears next to the key. Rotate the HDG knob and verify the heading bug, displayed on PFD1 compass card, moves and the flight director follows.

(h) Press the HDG knob and verify the heading bug centers.

(i) Press the YD key and verify the white illumination appears next to the key.

(j) Press the VS key and verify the white illumination appears next to the key.

(k) Press the FLC key and verify the white illumination appears next to the key.

(l) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(19) GWX 68 Test

WARNING: BEFORE ENERGIZING THE EQUIPMENT, BE SURE MICROWAVE RADIATION SAFETY PRECAUTIONS INCLUDING BOTH FUEL AND PERSONNEL SAFETY CONSIDERATIONS HAVE BEEN OBSERVED. THESE INCLUDE CLEARING ALL PERSONNEL TO AN AREA BEYOND THE MAXIMUM PERMISSIBLE EXPOSURE LEVEL (MPEL) BOUNDARY. THE MPEL FOR THE GWX 68 IS 11 FEET.

Operation of the GWX 68 Weather Radar is accomplished using the GCU 476.

(a) Start the G1000 in normal mode.

(b) On the MFD, turn the large FMS knob to select the Map Page Group on the MFD, and then turn the small FMS knob to select the Weather Radar page.

(c) Select the MODE softkey, then select the STANDBY softkey to initiate the one minute warm-up period. Verify the radar enters the standby mode after the warm-up is complete.

(d) Press the MFD FMS knob to activate the cursor in the TILT field then turn the small FMS knob to select the desired antenna tilt angle. Press the MFD FMS knob to remove the cursor.

(e) Select the MODE softkey, then select the WEATHER softkey to energize the radar array. A CAUTION window will come up. After complying with the CAUTION, press YES to continue. Press the BACK softkey to show weather radar scan and gain control softkeys.

(f) Select the GAIN softkey to activate the cursor in the ‘GAIN’ field. Turn the small FMS knob to adjust the gain to a non-calibrated level then press ENTER on the MFD. Verify the gain setting is visible in the gain field as a movable horizontal bar in a flashing box with a line pointer reference depicting the calibrated position. Press the FMS knob to remove the cursor.

(g) Select the VERTICAL softkey, repeat paragraph (f).

(h) Select the GAIN softkey again to recalibrate the gain. Verify ‘CALIBRATED’ is displayed in the ‘GAIN’ field. De-select the GAIN softkey.

(i) Select Horizon, Mode, and Ground softkeys sequentially. Verify scanning of and depiction of surrounding ground features. Verify Tilt, Bearing, Sector Scan, and Gain can be adjusted from the GROUND scan mode display via the FMS knob and RANGE knob, as appropriate.

(j) Select the MODE softkey, then select the STAB ON softkey to activate antenna stabilization. Verify that STAB ON is shown in the upper right corner of the weather radar display. Select the STAB OFF softkey to deactivate and verify that STAB OFF is shown in the upper right corner of the weather radar display.

(k) Select the OFF softkey at the conclusion of testing to turn off the weather radar.

(l) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(20) GAE 43 Calibration

(PIR-PPS55027, Rev. New.)

PA-46-350P only.

The GAE 43 altitude encoder is used to generate cabin altitude and rate information for the G1000 system.

(a) Required Equipment

Barfield Pitot Static Test Set, P/N 101-00164, (or equivalent) with male 1/8-27 NPT fitting plumbed to end of static line.

(b) Requirements

Each 24 months and whenever the GAE 43 encoder unit is replaced, calibrate the new unit before installation per procedure, below.

(c) Procedure

CAUTION: USE A BACKUP WRENCH ON THE FEMALE 1/8-27 NPT PIPE FITTING ON THE REAR OF THE GAE-43 WHEN ATTACHING THE STATIC LINE FROM THE TEST UNIT TO AVOID DAMAGING THE GAE-43 UNIT.

1) Wrap the male end of the test set static line in Teflon tape (MMM No. 547 or equivalent) and connect to the female 1/8-27 NPT pipe fitting on the rear of the GAE 43 (see "Figure 19").

2) Apply ground power and start the MFD and PFDs in normal mode.

3) Set the test set pressure altitude to 25,000 ft.

4) Adjust the GAE 43 such that the cabin pressure altitude indicated on the MFD is within 20 ft of the altitude set on the tester, as follows:

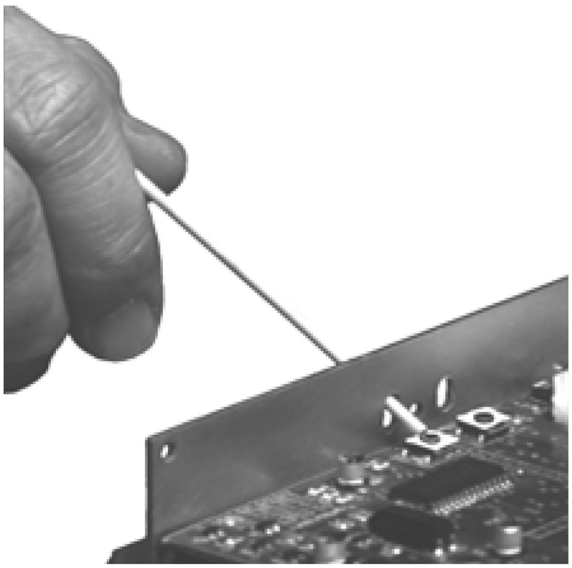
a) To increase the cabin pressure altitude indicated on the MFD:

Press the increase button behind slot A on the rear of the GAE 43, using a non-conductive tool ("Figure 50").

b) To decrease the cabin pressure altitude indicated on the MFD:

Press the decrease button behind slot B on the rear of the GAE 43, using a non-conductive tool ("Figure 50").

5) Set the test set pressure altitude to -900 ft. If, due to ambient atmospheric conditions, the test set is not capable of reaching -900 ft altitude, use the lowest negative altitude achievable on the test set as a calibration point.



Adjusting GAE 43
Figure 50

NOTE: The MFD altitude indicated in step 6 should be calibrated to ± 20 ft of the actual achievable altitude applied via the test set, if it differs from -900 ft.

6) Adjust the GAE 43 such that the cabin pressure altitude indicated on the MFD is within 20 ft of the altitude set on the tester, per step 4).

7) Set the test set pressure altitude to 5,000 ft.

8) Adjust the GAE-43 such that the cabin pressure altitude indicated on the MFD is within 20 ft of the altitude set on the tester, per step 4).

9) Repeat steps 7 and 8 at 10,000 ft and 12,500 ft test set pressure altitudes.

10) Remove power from the G1000 system. Disconnect the test set static line from the rear of the GAE-43 unit.

(21) Stormscope (WX-500) Functional Check

If installed.

(a) S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153.

1) Start the G1000 in the configuration mode.

2) Select the OTHER page group on the PFD. The stormscope page is shown by default.

3) Activate the cursor and highlight the DATA field. Select 'Config' and press the ENT key on the PFD.

4) Verify that the DATA window shows the following:

a)	Hdg: None: J3-1	Open
b)	J3-2	Open
c)	Hdg Valid Flag	N/A
d)	Flagsense	N/A
e)	Hdg Value	N/A
f)	Inhibit Line	Off
g)	Antenna Mount	Bottom
h)	J3-3	Open

5) Deactivate the cursor.

(b) S/N's 4636633, 4636652 and up.

1) Start the G1000 system in normal mode.

2) On the MFD, turn the large FMS knob to select the MAP group, then turn the small FMS knob to select the STORMSCOPE page.

3) During initialization, the WX-500 stormscope runs a series of self-tests to ensure that all functions are operating properly.

a) If no faults are detected continue to step 4.

b) If a fault is detected, consult the WX-500 Installation manual for troubleshooting assistance.

4) On the MFD, verify that it is possible to enter the following modes via softkeys: 360 degree, 120 degree, cell mode, strike mode.

5) Verify that the range can be changed from 25 NM to 200 NM via the range knob.

(22) TAWS Functional Check

If installed.

(a) Basic

1) Ensure that matching terrain data base cards are installed in the bottom card slots in the PFD and MFD, and the aircraft has a GPS position.

2) Select the TAWS-B page (last page in the MAP group).

3) Verify that the title at the top of the page reads "MAP - "TAWS-B". If TAWS has not been enabled, the title will read "MAP - TERRAIN PROXIMITY" or "MAP - TERRAIN".

4) Press the MENU button and select "Test TAWS" from the pop-up menu and press ENT.

5) After the TAWS test has completed, verify that "TAWS System Test Okay" is annunciated over the audio system.

(b) Detailed

In S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153 a more detailed check may be performed as follows:

1) With the G1000 in Normal Mode, use the GCU FMS knob to select the MAP group and TAWS page on the MFD.
2) Verify that the title at the top of the page reads “MAP – TAWS”.

NOTE: If TAWS has not been enabled, the title will read “MAP – TERRAIN PROXIMITY” or “MAP – TERRAIN”. Refer to TAWS Configuration for configuring TAWS.

3) Press the GCU MENU button and select “Test TAWS” from the pop-up menu. Verify TAWS test annunciation is displayed on the MFD and both PFDs.
4) After the TAWS test has completed, verify that “TAWS System Test Okay” is heard over the cockpit speaker.
5) Press the GCU MENU button again and select “Inhibit TAWS” from the pop-up menu and press ENT on the GCU. Verify “TAWS INHB” is displayed on PFD1 and PFD2.
6) Press the GCU MENU button again and select “Enable TAWS” from the prop-up menu and press ENT on the GCU. Verify the “TAWS INHB” annunciation on the PFDs has extinguished.
7) With a GPS position acquired, shield or disconnect the GPS antennas to remove the GPS signal. Verify “DR” shows on the MFD and the “TAWS N/A” and “DR” annunciations show on the PFDs.
8) Reconnect or remove the shield from the GPS antennas, and verify the MFD “DR” indication and PFD “TAWS N/A” and “DR” annunciations are removed once the GPS satellites are acquired.
9) Pull PFD1 PRI and PFD1 SEC circuit breakers. Re-power PFD1 in configuration mode, and use the PFD1 FMS knob to select the Audio Alert Configuration page.
10) Ensure cockpit speaker is selected ON. Use the PFD1 FMS knob to highlight each of the following messages then select PLAY. Verify each of the following audio messages can be played:
    - PDA – Caution: Too Low Terrain
    - EDR – Caution: Sink Rate
    - EDR – Warning: Pull Up
    - NCR – Caution: Don’t Sink
    - VCO – Caution: Five Hundred
    - RTC – Caution: Caution, Terrain (2X)
    - RTC – Warning: Terrain (2X), Pull Up (2X)
    - ROC – Caution: Caution, Obstacle (2X)
    - ROC – Warning: Obstacle (2X), Pull Up (2X)
    - ITI – Caution: Terrain Ahead (2X)
    - ITI – Warning: Terrain Ahead, Pull Up (2X)
    - IOI – Caution: Obstacle Ahead (2X)
    - IOI – Warning: Obstacle Ahead, Pull Up (2X)
11) Pull the PFD1 PRI and PFD1 SEC circuit breakers, and re-power PFD1 in normal operating mode.
12) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(23) ChartView Functional Check

If installed.

ChartView must be enabled using a ChartView Enable Card, and a current ChartView database. See "CHARTVIEW Unlock (Optional)" on page 342501160 for enabling procedures.

NOTE: The required ChartView databases are subscription-based and are to be procured by the aircraft owner directly from Jeppesen.

(a) Basic

1) Select the Airport Information page (first page in the WPT group).
2) Verify that the title at the top of the page reads “WPT- Airport Information”.
3) Enter KVRB as the waypoint and verify that the CHRT softkey is available.
4) Press the CHRT softkey and verify the electronic chart is viewable.

(b) Detailed

1) With the G1000 in Normal Mode, use the GCU FMS knob to select ‘AUX – System Status’ page and verify ChartView database cycle number is displayed in blue text and ChartView database is current.
2) From the Navigation Map page, press the SHW CHRT softkey and verify airport chart is displayed and the following softkeys are displayed:
   - CHRT OPT
   - CHRT
   - INFO
   - DP
   - STAR
   - APR
   - WX
   - NOTAM
   - GO BACK
3) Press CHRT OPT softkey and verify softkeys advance to the following level of softkeys:
   - ALL
   - HEADER
   - PLAN
   - PROFILE
   - MINIMUMS
   - FIT WDTH
   - FULL SCN
   - BACK
4) Return to the Navigation Map Page.
5) Press the GCU MENU key to display the PAGE MENU. Turn the GCU large FMS knob to scroll through the OPTIONS Menu to ‘Show Chart’. Press the GCU ENT key to display the chart and verify airport diagram is being displayed.
6) Press the GCU FMS knob to activate the cursor. Turn the GCU large FMS Knob to select the Airport Identifier Box.
7) Turn the GCU small and large FMS knob to enter the airport identifier for New Century airport (KIXD) then press the GCU ENT key to complete the airport selection.
8) Select the APR softkey. Turn the GCU large FMS Knob to select the Approach Box then turn the small FMS Knob to show the approach chart selection choices.

9) Turn either GCU FMS knob to scroll through the available charts and select a chart for viewing by pressing the GCU ENT key to complete the chart selection and verify the appropriate ChartView chart is displayed.

10) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(24) Terrain Proximity Check

If TAWS is not installed, in S/N's 4636633, 4636652 and up.

(a) Ensure that matching terrain data base cards are installed in the bottom card slots in the PFDs and MFD and the aircraft has a GPS position.

(b) On the MFD, select the MAP group, TERRAIN PROXIMITY page by rotating the large FMS knob to change groups and the small FMS knob to change pages.

(c) Verify the title at the top of the page reads "MAP – TERRAIN PROXIMITY".

(d) Verify that the map is colored to show terrain and no failures are annunciated on the page.

(25) Synthetic Vision Check

If installed.

NOTE: You will need a valid GPS signal.

(a) Ensure that matching terrain data base cards are installed in the bottom card slots in the PFD and MFD, and the aircraft has a GPS position.

(b) Press the PFD softkey on the PFD, then press the SYN VIS softkey.

(c) Press the SYN TERR softkey.

(d) Verify the traditional blue/brown attitude depiction is replaced with the Synthetic Vision rendering within 2–3 minutes.

(26) SafeTaxi Functional Check (If installed.)

The maximum map ranges for enhanced detail are configurable by the flight crew. When zoomed in close enough to show the airport detail, the map reveals runways with numbers, taxiways with identifying letters/numbers, and airport landmarks including ramps, buildings, control towers, and other prominent features. Resolution is greater at lower map ranges. When the aircraft location is within the screen boundary, including within SafeTaxi ranges, an airplane symbol is shown on any of the navigation map views for enhanced position awareness. Any map page that displays the navigation view can also show the SafeTaxi airport layout, within the maximum configured range. The following is a list of pages where the SafeTaxi feature can be seen:

- Navigation Map Page
- Inset map
- Weather Datalink Page
- Airport Information Page
- Intersection Information Page
- NDB Information Page
- VOR Information Page
- User Waypoint Information Page
- Trip Planning Page
- Nearest Pages

(a) Use the FMS knob on the GCU to select the AUX – System Status page and select DBASE softkey. Use the small FMS knob to scroll to CHART category and verify "ChartView" is displayed in blue text adjacent to "CHART".

(b) Verify the Chartview database 'REGION', 'CYCLE' number, 'EFFECTIVE', 'EXPIRES', and 'DISABLES' dates of the subscription appear in blue text.

(c) Use the FMS knob on the GCU to select MAP - Navigation Map page.

(d) On the GCU, press MENU. With Map Setup highlighted, press ENT on the GCU. Rotate the small GCU FMS knob to select the Aviation group and press the ENT key on GCU.

(e) Turn the GCU large FMS Knob to scroll through the Aviation Group options to ‘SAFETAXI’.

(f) Turn the GCU small FMS Knob to display the range of distances.

(g) Turn the GCU FMS Knob to select 5000ft as the distance for maximum SafeTaxi display range, and then press the GCU ENT key to complete the selection.

(h) Using the GCU range knob, select a range of 5000ft or less. Verify SafeTaxi display represents the current aircraft location and the airport layout. If no other service is to be performed, continue to the return-to-service.

(27) KTA-870 Checkout

If installed, S/N’s 4636460, 4636463–4636651, less 4636481 and 4636633; 4692134 and up, less 4692141, 4692149, 4692153.

(a) Start the G1000 system in normal mode and select the TRAFFIC MAP page on the MFD

(b) Verify that the STANDBY, NORMAL, TEST (in Standby mode) and ALT MODE softkeys are available on the bottom of the display. Verify that a TAS Status (not TAS FAIL) is displayed in the upper left corner of the traffic map. Verify that a yellow NO DATA message is not displayed in the center of the map. If TIS is displayed instead of TAS, refer to "KTA-870 Configuration" on page 342501159 to configure the KTA 870 option.

(c) Press the NORMAL softkey and verify that OPERATING is displayed in the upper left corner of the traffic map.

(d) Press the STANDBY softkey and verify that STANDBY is displayed in the upper left corner of the traffic map.

(e) Press the TEST softkey and verify that TEST is displayed in the upper left corner of the traffic map and that a traffic test pattern is displayed. Verify that “TAS System Test Ok” annunciation is heard over the cockpit speaker at the conclusion of the test.

(f) Open the TAS circuit breaker and verify that NO DATA is displayed at the center of the traffic map after several seconds.

(g) Close the TAS circuit breaker and verity that NO DATA is removed from the traffic map.

(28) GTS 825

If Installed, S/N's 4636633, 4636652 and up.

(a) System Check

1) Start the G1000 system in normal mode and select the TRAFFIC MAP page on the MFD MAP group.

2) Verify that the STANDBY, OPERATE, TEST (in Standby mode) and ALT MODE softkeys are available on the bottom of the display. Verify that a TAS Status (not TAS FAIL) is displayed in the upper left corner of the traffic map. Verify that a yellow NO DATA message is not displayed in the center of the map. If TIS is displayed instead of TAS, refer to "Optional Equipment Configuration" on page 342501154, above, to install the GTS 825 option.

3) Press the OPERATE softkey and verify that OPERATING is displayed in the upper left corner of the traffic map.

4) Press the STANDBY softkey and verify that STANDBY is displayed in the upper left corner of the traffic map.

5) Set range to 6NM or greater.

6) Press the TEST softkey and verify that TEST is displayed in the upper left corner of the traffic map and that a traffic test pattern is displayed. Verify that "TAS System Test Passed" aural is heard over the cockpit speaker at the conclusion of the test.

7) Open the TAS circuit breaker and verify that NO DATA is displayed at the center of the traffic map after several seconds.

8) Close the TAS circuit breaker and verity that NO DATA is removed from the traffic map.

(b) Function Test

Perform the following test to verify GTS 825 operational and surveillance functions. Use a ramp tester such as a TIC TR220 or equivalent to perform the tests.

1) To select a scenario that will properly converge and intercept the GTS 825, the GTS 825 must be in ground test mode. To enable ground test mode, the aircraft must be on the ground and the GTS 825 must be in normal system mode and in standby.

2) Position the test set directional antenna with a clear line of sight to the GTS 825 antenna at 90 degrees. With the GTS 825 powered up and in standby mode indicated on the CDTI, cycle the GTS 825 to 'Operate.'

3) Select the following:

a) Set the intruder type as ATCRBS.

b) Intruder Start Distance: 10 nm.

c) Intruder Start Altitude: 50,000 ft.

d) Vertical Speed: 0 fpm.

e) Velocity: 360 kts.

4) Initiate the intruder scenario and observe the following:

a) Traffic should be acquired at approximately 10 NM at 90 degree bearing and co-altitude. Observe intruder closes on own aircraft at a rate of 0.1 NM/sec.

b) The intruder should transition from Other Traffic (displayed as an open diamond with 00 displayed above), to proximate traffic (displayed as a filled white diamond with 00 displayed above), to a Traffic Advisory (TA) alarm.

c) The appropriate TA symbology (yellow filled circle with 00 displayed above, and an audio annunciation of "Traffic! 3 O'clock! At Altitude! 3 Miles!"), displayed when the intruder approaches within 3 NM.

(29) GSR 56 Checkout

If Installed, S/N's 4636633, 4636652 and up.

NOTE: This procedure verifies correct installation of the GSR 56.

See "Chart 27" on page 342501119, if the Iridium Satellite Radio/Datalink requires activation.

(a) On the MFD navigate to AUX group, SATELLITE PHONE page using the FMS knobs.

(b) Press the DIAL softkey.

(c) Use the softkeys or FMS knobs to enter “1911135” and press ENT to confirm.

(d) Press ENT to call.

(e) If both parties can hear each other, then the installation has been verified. Tell the representative that you are testing the GSR 56 installation and that the unit is active.

NOTE: If placed on hold when calling Garmin Flight Data Services, you may call anyone for a brief verification phone call. Normal charges will apply.

(30) ADF Adjustment and Checkout

If installed. Select an appropriate local NDB (Station 1) and determine the station's bearing from the test location.

(a) Quadrantal Error Adjustment

S/N's 4636633, 4636652 and up.

1) Position aircraft on ramp in an area that is clear of surrounding buildings and other obstructions that may block or reflect radio signals.

2) Select ADF as the bearing source on the #1 PFD.

3) Select ADF mode on ADF receiver.

4) Tune the ADF to Station 1.

5) Orient aircraft such that the bearing indicator points to 0 degrees (aircraft is heading directly to Station 1). Note the heading (Direction A).

6) Using the heading indicator, rotate the aircraft 45 degrees to the right (Direction A plus 45 degrees).

7) ADF indicator should show 315 ±3 degrees.

8) Using the heading indicator, rotate the aircraft 90 degrees to the left (Direction A minus 45 degrees).

9) ADF indicator should show 45 ±3 degrees.

NOTE: The following steps are not required if the quadrantal error is within tolerance.

10) Adjust the variable resistor R60 on the rear side of the RMI-Converter to correct the quadrantal error.

11) Recheck the relative bearings and readjust the QE variable resistor as necessary to split the errors at the quadrantal points and obtain the lowest possible average error.

(b) Checkout

1) Tune the ADF to Station 1 and verify the bearing is with +/- 15° of the known bearing to Station 1 and the Morse Code Identifier is clear and accurate.

2) Verify the ADF audio is available on the GMA 347/350 audio panel.

(c) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(31) DME Functional Check

If installed.

(a) Perform systems check with appropriate DME test equipment (i.e., ATC 600, etc.) and verify the DME is tuned by NAV 1 and NAV 2.

(b) Verify the distance is accurate.

(c) Verify the DME audio is available on the GMA 347/350 audio panel.

(d) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(32) GTX 33/33D (ES) Testing

(a) General

Operation of the GTX 33/33D (ES) Mode S transponder is accomplished using PFD1, PFD2 or the MFD. Refer to G1000 Cockpit Reference Guide, for basic operation.

The integrated transponder/altitude reporting system must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) §§ 91.411 and 91.413, every 24 calendar months, or any time the transponder is removed. This test requires the use of a Mode S ramp generator. Specific instructions for operating the ramp tester are contained in the applicable operator's manual. Refer to Title 14 CFR Part 43.

(b) GTX 33 ES (Extended Squitter) Checkout

If installed.

1) Press the XPDR softkey on the #1 PFD. Then press the XPDR1 softkey.
2) Verify that the ADS-B TX softkey is available and can be toggled (softkey 10).
3) If dual Extended Squitter is installed, press XPDR2 softkey and verify that the ADS-B TX softkey is available and can be toggled (softkey 10).

(c) If no other service is to be performed, see "Return to Service Procedure" on page 342501213.

(33) Display System Checkout

NOTE: Items in [brackets] below apply to S/N's 4636633, 4636652 and up.

(a) NAV Source Selection Tests

NOTE: The CDI will not be depicted if the VOR antenna is not receiving a valid signal.

NOTE: For GPS, the word that is displayed with the GPS (i.e., GPS Term) depends on the airplane's phase of flight. Either GPS TERM or GPS ENR is acceptable.

1) Start these tests with all avionics powered.
2) Press the Direct-To-Key (~~D~~>).
3) Enter a Direct-To waypoint. [Press ENT to accept waypoint.]
4) Enter 117.3 MHz (TRV VOR) or an appropriate local VOR frequency into the active NAV 1 frequency box on #1 PFD.
5) Enter 110.0 MHz (MLB VOR) or an appropriate local VOR frequency into the active NAV 2 frequency box on #1 PFD.
6) On MFD, AUX Group page 5, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight CDI in the Synchronization window. Turn the small FMS knob to turn synchronization ON. Press the FMS knob to deactivate the cursor.
7) Verify that PFD1 and PFD2 synchronize (i.e., PFD1 & PFD2 match for the CDI).
8) Press the CDI softkey on PFD1. Verify the NAV source on PFD1 and PFD2 changes from GPS TERM to VOR1.
9) Press the CDI softkey on PFD1. Verify the NAV source on PFD1 and PFD2 changes from VOR1 TO VOR2.

10) Press the CDI softkey on PFD1. Verify the NAV source on PFD1 and PFD2 changes from VOR2 TO GPS TERM.
11) Repeat Step 6), but turn the synchronization OFF.
12) By pressing the CDI softkey on PFD1, cycle the NAV source on PFD1 from GPS TERM to VOR1 TO VOR2 TO GPS TERM. Verify that only the NAV source depicted on PFD1 changes from GPS TERM to VOR1 TO VOR2 TO GPS TERM.
13) By pressing the CDI softkey on PFD2, cycle the NAV source on PFD2 from GPS TERM to VOR1 TO VOR2 TO GPS TERM. Verify that only the NAV source depicted on PFD2 changes from GPS TERM to VOR1 TO VOR2 TO GPS TERM.

(b) Bearing Pointers and Information Windows Tests

NOTE: The bearing pointer will not be depicted if it is not receiving a valid signal.

1) Start these tests with all avionics powered.
2) Press the Direct-To-Key (D→).
3) Enter a Direct-To waypoint. [Press ENT to accept waypoint.]
4) Enter 117.3 MHz (TRV VOR) or an appropriate local VOR frequency into the NAV1 frequency box on PFD1. Enter 110.0 MHz (MLB VOR) or an appropriate local VOR into the NAV2 frequency box on PFD1.
5) [Press ADF/DME of PFD1.] Enter 257 kHz (SQT NDB) or an appropriate ADF frequency into the ADF receiver (if installed).
6) Press the PFD softkey on PFD1 to make the BRG1 and BRG2 softkey selections visible.
7) Verify that the BRG1 softkey can be pressed to sequentially cycle through NAV1, GPS, and ADF (if installed).
8) Verify that the BRG2 softkey can be pressed to sequentially cycle through NAV2, GPS, and ADF (if installed).
9) Repeat Steps 6)–8) for PFD2.

(c) BARO Selection Tests

1) On MFD AUX Group page 5, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization ON. Press the FMS knob to deactivate the cursor.
2) Rotate the BARO knob on PFD1 and verify that both PFD's BARO values change.
3) Rotate the BARO knob on PFD2 and verify that both PFD's BARO values change.
4) Set the current altimeter setting and verify that the both PFD's indicate within ±30 ft. of local field elevation.
5) On MFD AUX Group page 5, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization OFF. Press the FMS knob to deactivate the cursor.
6) Rotate the BARO knob on PFD1 and verify that only PFD1 BARO values change and a delta indication of at least 0.3 in. Hg [0.01 in. Hg] turns the indication yellow.
7) Rotate the BARO knob on PFD2 and verify that only PFD2 BARO values change and a delta indication of at least 0.3 in. Hg [0.01 in. Hg] turns the indication yellow.
8) On MFD AUX Group page 5, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

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O. Return to Service Procedure

WARNING: SOFTWARE VERSIONS MUST BE CHECKED AND MATCHED AGAINST THE LISTED VERSIONS IN THE APPROPRIATE REQUIRED EQUIPMENT LIST (REL), SEE "CHART 39" ON PAGE 342501229. SOFTWARE CONFIGURATION IS A CRITICAL PART OF THE G1000 OPERATION AND MUST BE VERIFIED BEFORE RETURNING AN AIRCRAFT TO SERVICE.

(1) Software Verification

After reinstalling any G1000 LRU, use the checkoff list in "Chart 38" on page 342501228 to verify correct LRU software part numbers and versions against the current numbers and versions shown in the appropriate Required Equipment List (REL) ("Chart 39" on page 342501229).

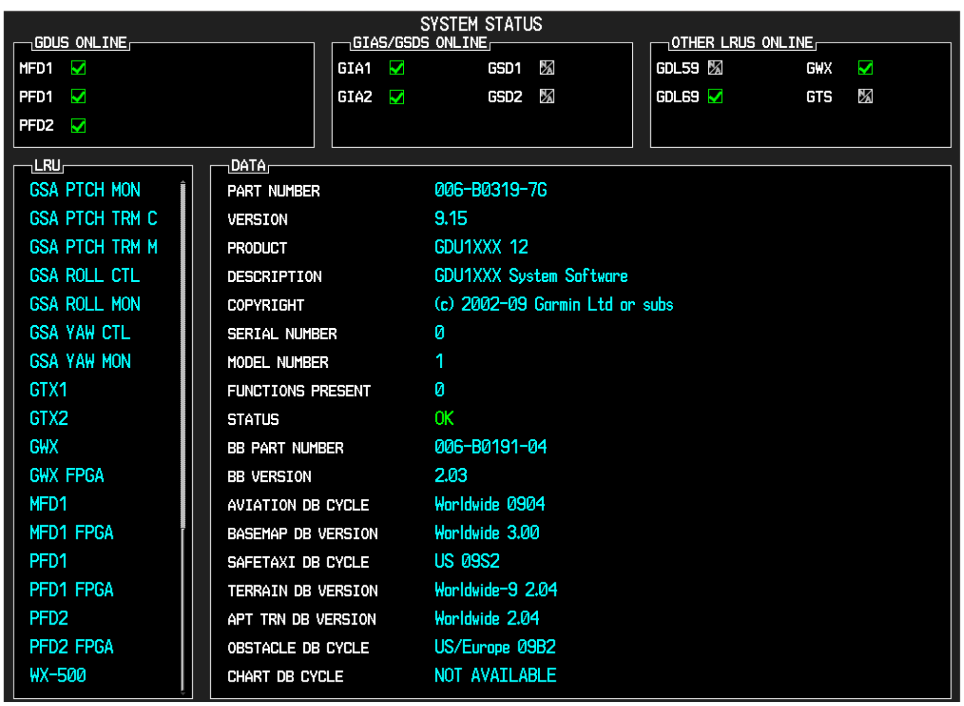
(a) Start the G1000 system in configuration mode.

(b) The PFD System Status page (see "Figure 51") displays a list of LRUs in the LRU window.

(c) Activate the cursor and highlight the LRU window.

(d) Use the FMS knob to scroll through the list, selecting each LRU in turn.

(e) The software part number and version is displayed in the DATA window. Compare this to the current numbers and versions shown in the appropriate Required Equipment List (REL) ("Chart 39" on page 342501229).



System Status Page (Configuration Mode)
Figure 51

NOTE: The following Failure Tests use various secondary communications paths to ensure that the desired backup paths are in place.

(2) GIA Failure Test

NOTE: Items in {braces} below apply to S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153; items in [brackets] below apply to S/N's 4636633, 4636652 and up.

(a) Start these tests with all avionics powered, valid GPS reception, valid NAV signal.

(b) Engage the autopilot by pressing the AP key on AFCS mode controller.

(c) If XFR button is not pointing left, couple FD to PFD1 by pressing the XFR button on the AFCS mode controller.

(d) Remove power from GIA 1 by pulling the INTEG AV 1 and COM 1 circuit breakers.

(e) Verify the following:

1) Autopilot and yaw damper disconnect with a continuous AP disconnect tone.

2) The FD bars are removed.

3) A red-boxed AFCS Annunciation appears on the PFD.

(f) Acknowledge the tone by pressing the AP DISC TRIM INTER switch.

(g) Verify the following system messages:

1) AHRS1 GPS – AHRS1 USING BACKUP GPS SOURCE

2) FAILED PATH

3) AHRS2 GPS – AHRS2 NOT RECEIVING BACKUP GPS INFORMATION

4) XPDR1 FAIL (If dual GTX 33 option is installed)

5) [GMA1 FAIL]

(h) Verify the following flag invalid:

1) COM/NAV 1 field

2) NAV 1 CDI loses deviation bar.

3) XPDR 1 FAIL (If dual GTX 33 option is installed).

4) Fuel Quantity

(i) Verify BOTH ON GPS2 Sensor Reversion annunciation on the PFDs.

(j) Restore power to GIA1 by resetting the INTEG AV 1 and COM 1 circuit breakers.

(k) Verify the following:

1) All invalid flags, system messages, and annunciations are removed. (#1 GPS will take time to reinitialize).

2) Autopilot passes the Preflight Test (PFT).

(l) Engage the autopilot by pressing the AP key on AFCS mode controller.

(m) Verify FD is coupled to PFD1 as indicated by a left pointing arrow on the AFCS mode controller next to the XFR button. If necessary, couple FD to PFD1 by pressing the XFR button.

(n) Remove power from GIA 2 by pulling the INTEG AV 2 and COM 2 circuit breakers.

(o) Verify the following:

1) Autopilot and yaw damper disconnect with a continuous AP disconnect tone.

2) The FD bars remain in view.

3) A red-boxed AFCS Annunciation appears on the PFD.

(p) Acknowledge the tone by pressing the AP DISC TRIM INTER switch.

(q) Verify the following system messages:
  1) AHRS2 GPS – AHRS1 USING BACKUP GPS SOURCE (if installed)
  2) STRMSCP FAIL (if installed)
  3) AHRS1 GPS – AHRS1 NOT RECEIVING BACKUP GPS INFORMATION.
  4) XPDR2 FAIL (If dual GTX 33 option is installed)
  5) {TRAFFIC FAIL}
  6) [FAILED PATH]
  7) [CO DET FAIL]
(r) Verify the following flag invalid:
  1) COM/NAV 2 field
  2) NAV 2 CDI loses deviation bar.
  3) Stormscope (on MFD Stormscope Map page) via LIGHTNING FAILED yellow text (if installed).
  4) {XPDR2 FAIL (If dual GTX 33 option is installed).}
  5) {Traffic (on MFD Traffic page) via NO DATA text, no traffic symbols, and FAIL displayed in the upper left corner.}
(s) Verify BOTH ON GPS1 Sensor Reversion annunciation on the PFDs.
(t) Restore power to GIA2 by resetting the INTEG AV 2 and COM 2 circuit breakers.
(u) Verify the following:
  1) All invalid flags, system messages, and annunciations are removed. (#2 GPS will take time to reinitialize).
  2) Autopilot passes the Preflight Test (PFT).
(v) Remove power from GIA 1 and GIA 2 by pulling the INTEG AV 1, INTEG AV 2, COM 1 and COM 2 circuit breakers. {For a dual GIA failure, only the following shall occur:}
(w) In S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153; verify the following:
  1) COM/NAV 1 and COM/NAV 2 fields flag invalid.
  2) GPS1 and GPS2 failures annunciated on PFD.
  3) NAV 1, 2 CDI loses deviation bar.
  4) XPDR field flags invalid (If dual GTX 33 option is not installed).
  5) XPDR 1 Fail field and XPDR 2 Fail field flags invalid (If dual GTX 33 option is installed).
  6) In PA-46-350P only, Cabin Alt, Diff Pressure, and Cabin Rate fields flag invalid on MFD.
  7) Engine Instrument field flags invalid on MFD.
  8) All AHRS and ADC fields remain valid.

(x) In S/N's 4636633, 4636652 and up

1) Verify the following system messages:
   a) GPS NAV LOST
   b) TRAFFIC FAIL
   c) XPDR1 FAIL
   d) XPDR2 FAIL (if dual GTX option is installed)
   e) GMA1 FAIL
   f) TRN AUD FAIL (If TAWS is installed)
2) Verify the following flags invalid:
   a) COM/NAV 1 field
   b) COM/NAV 2 field
   c) XPDR field(s)
   d) Engine Instruments
   e) Fuel Quantity
   f) Cabin Alt, Diff Pressure, and Cabin Rate
3) Verify the following:
   a) All AHRS and ADC fields remain valid.
   b) NAV 1 & NAV 2 CDI lose deviation bar.
   c) "GEAR SYS" CAS warning.
   d) "CPCS FAIL" CAS caution ("CPCS FAULT" may also annunciate)

(3) GPS Failure Test

NOTE: Items in [brackets] below apply to S/N's 4636633, 4636652 and up.

(a) Start these tests with all avionics powered, Valid GPS reception.

(b) [Press Direct-To-Key (~~D~~>). Enter a Direct-To waypoint.]

(c) Place a shroud over the GPS 1 antenna to prevent signal reception. Verify loss of GPS 1 signal on MFD AUX group, GPS STATUS page. The following should remain valid on the PFD throughout the procedure.
1) Attitude
2) Heading
3) Airspeed
4) Altitude
5) Vertical Speed
6) OAT
7) GPS CDI remains valid on PFD.

(d) Remove shroud from the GPS 1 antenna.

(e) Place a shroud over the GPS 2 antenna to prevent signal reception. Verify loss of GPS 2 signal on MFD AUX group, GPS STATUS page. The following should remain valid on the PFD throughout the procedure.
1) Attitude
2) Heading
3) Airspeed
4) Altitude
5) Vertical Speed

6) OAT
7) GPS CDI remains valid on PFD.

(f) Leave the shroud over the GPS 2 antenna and replace the shroud over the GPS 1 antenna. Verify loss of GPS 1 and 2 signals on MFD AUX page 4. The following should remain valid on the PFD throughout the procedure.

1) Attitude
2) Heading
3) Airspeed
4) Altitude
5) Vertical Speed
6) OAT
7) Loss of GPS integrity annunciated [NO GPS POSITION] on the PFD.

(4) Display Failure Test

NOTE: Items in {braces} below apply to S/N's 4636460, 4636463–4636651, less 4636481, 4636633; and 4692134 and up, less 4692141, 4692149, 4692153; items in [brackets] below apply to S/N's 4636633, 4636652 and up.

(a) Start these tests with all avionics powered.
(b) {Select reversionary mode on #1 and #2 PFDs.}
(c) Select PFD on the GCU 476 Keypad. Verify FD is coupled to PFD1 by a left pointing arrow on the AFCS mode controller next to the XFR button.
(d) Engage the autopilot by pressing the AP key on the AFCS mode controller.
(e) Remove power from the MFD by pulling the MFD circuit breaker.
(f) Verify the following system messages on the PFDs:

1) XTALK ERROR
2) GWX FAIL
3) [FAILED PATH]
4) MFD1 VOLTAGE (may or may not appear depending on power failure rate vs. MFD update rate)
5) STRMSCP FAIL (if installed)
6) XPDR2 FAIL (if installed)
7) {TRAFFIC FAIL}
8) [CO DET FAIL]

(g) Verify the following:

1) COM/NAV 2 fields flag invalid.
2) ADF indication (BRG1 & BRG2) flag invalid (if option is installed.
3) DME indication flags invalid (if option is installed)
4) GMC 710 Autopilot controller remains functional while XFR is to PFD #1.
5) PFDs switch to reversion mode.
6) Attitude and Heading remain valid from AHRS.
7) Airspeed, Altitude, Vertical Speed and OAT remain valid.
8) Engine Instrumentation appears on PFDs.
9) GCU 476 Keypad functions on #1 PFD.

(h) Restore power to the MFD {and deselect reversionary mode on #1 PFD}. The displays should return to Normal operation.

(i) Activate pilot's DISPLAY BACKUP switch (reversionary mode on #1 PFD).

(j) Select MFD on the GCU 476 Keypad.

(k) Couple FD to PFD # 2 by pressing the XFR button on the AFCS controller.

(l) Remove power from the #1 PFD by pulling the PFD1 circuit breaker.

(m) Verify the following system messages are displayed on #2 PFD and the MFD:

1) XTALK ERROR
2) FAILED PATH
3) {GDL69 FAIL (If installed)}
4) XPDR1 FAIL [(if dual GTX option is installed)]
5) [TRAFFIC FAIL]
6) [GMA1 FAIL]

(n) Verify the following:

1) MFD [and PFD2] switch to reversion mode.
2) Attitude and Heading remain valid from AHRS.
3) Airspeed, Altitude, Vertical Speed and OAT remain valid.
4) Reversionary sensor BOTH ON GPS2 is displayed.
5) MFD retains engine instrumentation.
6) COM/NAV1 fields flag invalid.
7) GCU 476 Keypad functions on MFD.
8) GMC 710 Autopilot Controller remains functional while XFR is to #2 PFD.

(o) Restore power to the #1 PFD and deactivate the pilot's DISPLAY BACKUP switch. The displays should return to normal operation.

(p) Couple FD to PFD # 1 by pressing the XFR button on the AFCS controller.

(q) Select reversionary mode on #2 PFD by activating copilot's DISPLAY BACKUP switch.

(r) Remove power from the #2 PFD by pulling the PFD2 circuit breaker.

(s) Verify the following system messages are displayed on #1 PFD and the MFD:

1) XTALK ERROR
2) FAILED PATH
3) [GWX FAIL]

(t) Verify the following:

1) MFD [and #1 PFD] switch to reversion mode.
2) Attitude and Heading remain valid from AHRS.
3) Airspeed, Altitude, Vertical Speed and OAT remain valid.
4) MFD retains engine instrumentation.
5) [COM/NAV 2 fields flag invalid.]
6) GMC710 Autopilot controller remains functional while XFR is to PFD #1.

(u) Restore power to the #2 PFD and deactivate copilot's DISPLAY BACKUP switch. The displays should return to normal operation.

(5) AHRS Failure Tests

NOTE: Allow for the AHRS to realign between tests.

(a) Start these tests with all avionics powered.
(b) Engage the autopilot by pressing the AP key on the AFCS mode controller
(c) Remove power from the #1 AHRS by pulling the AHRS 1 circuit breaker.
(d) Verify the autopilot and yaw damper disconnect with a continuous AP disconnect tone.
(e) Acknowledge the tone by pressing the AP DISC TRIM INTER switch.
(f) Verify the following Comparator Annunciations are displayed on the PFDs:
1) HDG NO COMP
2) ROL NO COMP
3) PIT NO COMP
(g) Verify Heading, Pitch, and Roll flag invalid on #1 PFD.
(h) Press the SENSOR key on #1 PFD.
(i) Press the AHRS2 key on #1 PFD.
(j) Verify the following:
1) #2 AHRS parameters are displayed on #1 PFD without invalid flags.
2) BOTH ON AHRS2 sensor reversion annunciation is displayed on the PFDs.
3) White NO COMP Annunciations remain on the PFDs.
(k) Restore power to the #1 AHRS.
(l) Select AHRS1 key on #1 PFD.
(m) Verify normal operation, failure and Comparator Annunciations are removed.
(n) Repeat steps (b) through (m) for the #2 AHRS and #2 PFD.

(6) Cross-Side AHRS Test
(a) Start this test with all avionics powered.
(b) Press the Sensor key on each PFD.
(c) Press the AHRS2 Key on #1 PFD.
(d) Press the AHRS1 Key on #2 PFD.
(e) Verify the USING AHRS 2 annunciation on #1 PFD and USING AHRS 1 annunciation on #2 PFD.
(f) Press the AHRS1 Key on #1 PFD.
(g) Press the AHRS2 Key on #2 PFD.
(h) Verify AHRS annunciations are removed.

(7) Air Data Computer (ADC) Failure Tests
(a) Start these tests with all avionics powered.
(b) Engage the autopilot by pressing the AP key on AFCS mode controller.
(c) Remove power from the #1 ADC by pulling the #1 ADC circuit breaker.
(d) Verify the autopilot and yaw damper remain engaged. [Disconnect autopilot by pressing the AP button on the AFCS mode controller.]
(e) Verify the following Comparator Annunciations are displayed on the PFDs:
1) IAS NO COMP
2) ALT NO COMP

(f) Verify the following flag invalid #1 PFD.
  1) Altitude
  2) Indicated Airspeed
  3) Vertical Speed
  4) Static Air Temperature (OAT) and ISA
(g) {Verify Cabin Differential Pressure flags invalid on the MFD.}
(h) Press the SENSOR key on #1 PFD.
(i) Press the ADC2 key on #1 PFD.
(j) Verify the following:
  1) #2 ADC parameters are displayed on #1 PFD without invalid flags.
  2) BOTH ON ADC2 sensor reversion annunciation is displayed on the PFDs.
  3) White NO COMP Annunciations remain on the PFDs.
(k) Restore power to the #1 ADC.
(l) Select ADC1 key on #1 PFD.
(m) Verify normal operation, invalid flags and Comparator Annunciations are removed.
(n) Repeat steps (b) through (m) for the #2 ADC and #2 PFD.

(8) AHRS/Air Data Backup Path Test
(a) Start these tests with all avionics powered.
(b) Remove power from the #1 PFD by pulling the #1 PFD circuit breaker.
(c) On #2 PFD, press the SENSOR soft key and select ADC1 and AHRS1. This test verifies the secondary AHRS/ADC 1 paths. The following shall occur during this test:
  1) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.
  2) Engine Instrumentation remains valid except fuel quantity.
  3) COM/NAV 1 fields flag invalid.
(d) Restore power to the #1 PFD.
(e) Remove power from the MFD and GIA 1 by pulling the MFD and INTEG AV 1 circuit breakers.
(f) On #1 PFD, press the SENSOR softkey and select ADC1 and AHRS1. This test verifies the primary AHRS/ADC 1 paths. The following shall occur during this test:
  1) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.
  2) COM/NAV1 {and COM/NAV2} fields flag invalid.
(g) All engine instruments flag invalid.
(h) Restore power to the MFD and GIA 1.

(9) Display Testing

The G1000 system is tested while operating in the normal mode unless otherwise specified. If the system is in configuration mode, restart the displays by cycling the PFD1, PFD2 and MFD circuit breakers to start the display in the normal mode.

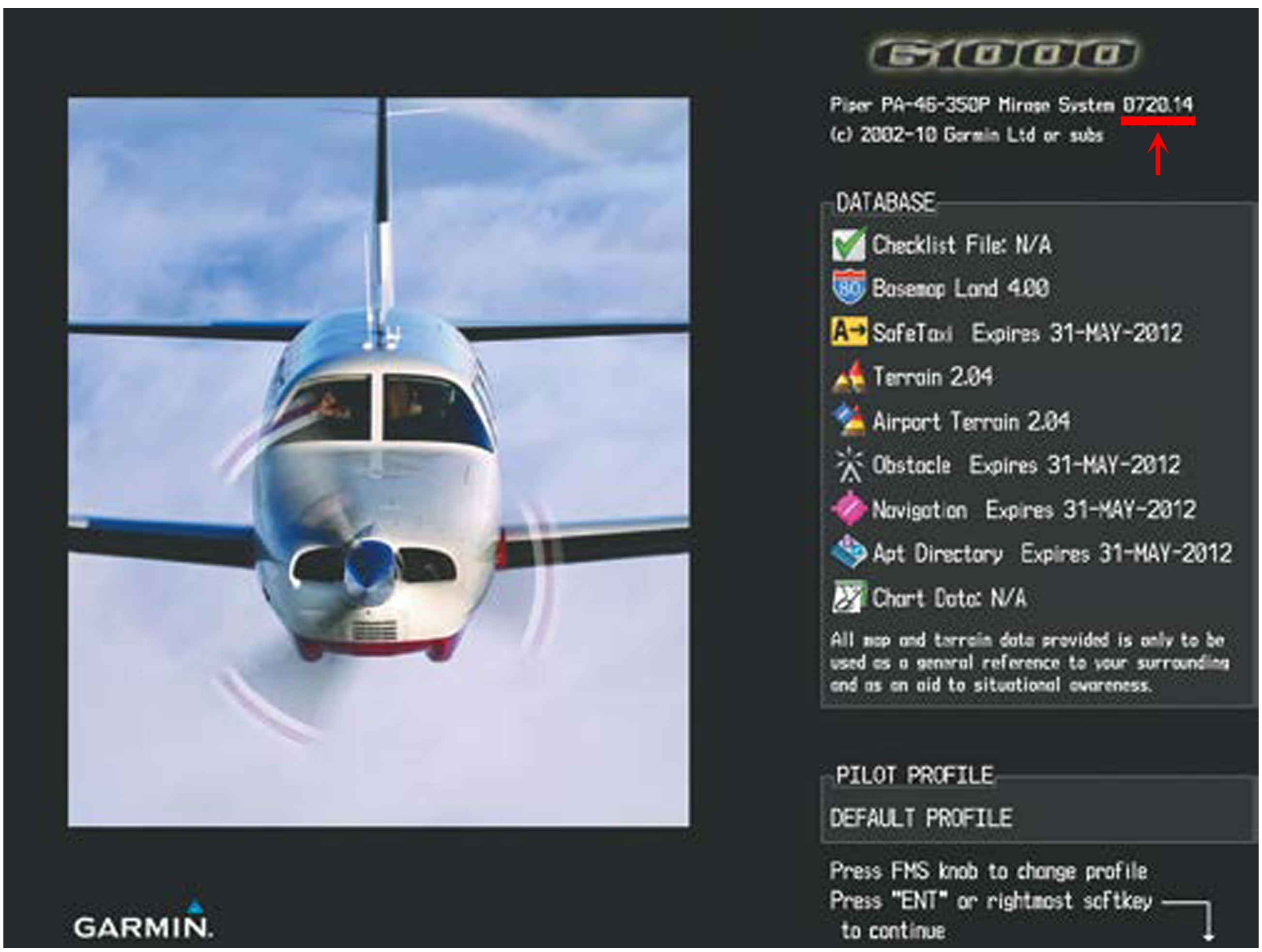
(a) Apply aircraft power. Observe the MFD power-up screen. Verify the display is formatted as shown in "Figure 52". This figure is a format reference, the illustration on the left may vary. Note database versions and system software represented in the upper right corner. Refer to the appropriate Required Equipment Lists (REL), "Chart 40", for the correct database and system software versions.

(b) The ‘System’ number reflected in the upper right hand corner is the System Software Version. It correlates to the G1000 SW Loader Card used to load the software to the system.

(c) Verify that the System Software Version is correct per the appropriate Required Equipment List (REL), "Chart 39" on page 342501229.

(d) Press the GCU ENT key to acknowledge the correct pilot profile on the MFD.

NOTE: The rightmost softkey on the MFD may also be used.



MFD Initial Power Up Page
Figure 52

(e) In the normal operating mode, data fields that are invalid have large red X's through them ("Figure 53"). A valid field does not display a red X ("Figure 54" on page 342501223). Allow the displays to initialize for approximately one minute. The GDC 74As requires a longer initialization period than do the other LRUs. During normal operation, this causes the airspeed, altitude, vertical speed, and OAT fields to be invalid during the first ~40–60 seconds of PFD power-up.

(f) Check that all COM/NAV fields are valid in the top corners of PFD1 and PFD2.

(g) Check that altitude, airspeed, vertical speed, TAS, and OAT fields are valid on PFD1 and PFD2.

(h) Press the SENSOR softkey on each PFD and switch between ADC1 and ADC2. Verify that data from both GDCs are valid on both displays.

(i) Press the SENSOR softkey on each PFD and switch between AHRS1 and AHRS2. Verify that data from both GRS 77s are valid on both displays.

(j) Check that engine instrument fields are valid on the MFD.

(k) Verify that no MANIFEST alert messages appear in the lower right corner (press the flashing ALERTS softkey to view alert messages). If any MANIFEST errors appear, the correct software to the related LRU must be loaded before proceeding.

PFD Power Up System Annunciations
Figure 53

PFD Operation
Figure 54

(10) Reversion Mode Check

(a) Push the DISPLAY BACKUP button/switch on the pilot's side. Verify that the pilot-side PFD and MFD displays enter reversion mode ("Figure 55"). MFD should have valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(b) De-activate pilot-side reversion mode. Verify PFD1 and MFD return to normal display modes.

(c) Repeat Step 1 using the DISPLAY BACKUP button/switch on the co-pilot's side. Ensure that PFD2 and MFD enter reversion mode and MFD displays valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(d) De-activate co-pilot's side reversion mode. Verify PFD2 and MFD return to normal display modes.

(e) Open PFD1 circuit breaker. Verify the PFD1 goes blank and MFD display remains in normal mode.

(f) Activate pilot-side reversion mode. Verify that the MFD displays enter reversion mode. MFD should have valid altitude, airspeed, vertical speed, COMM2, NAV2 and engine instruments.

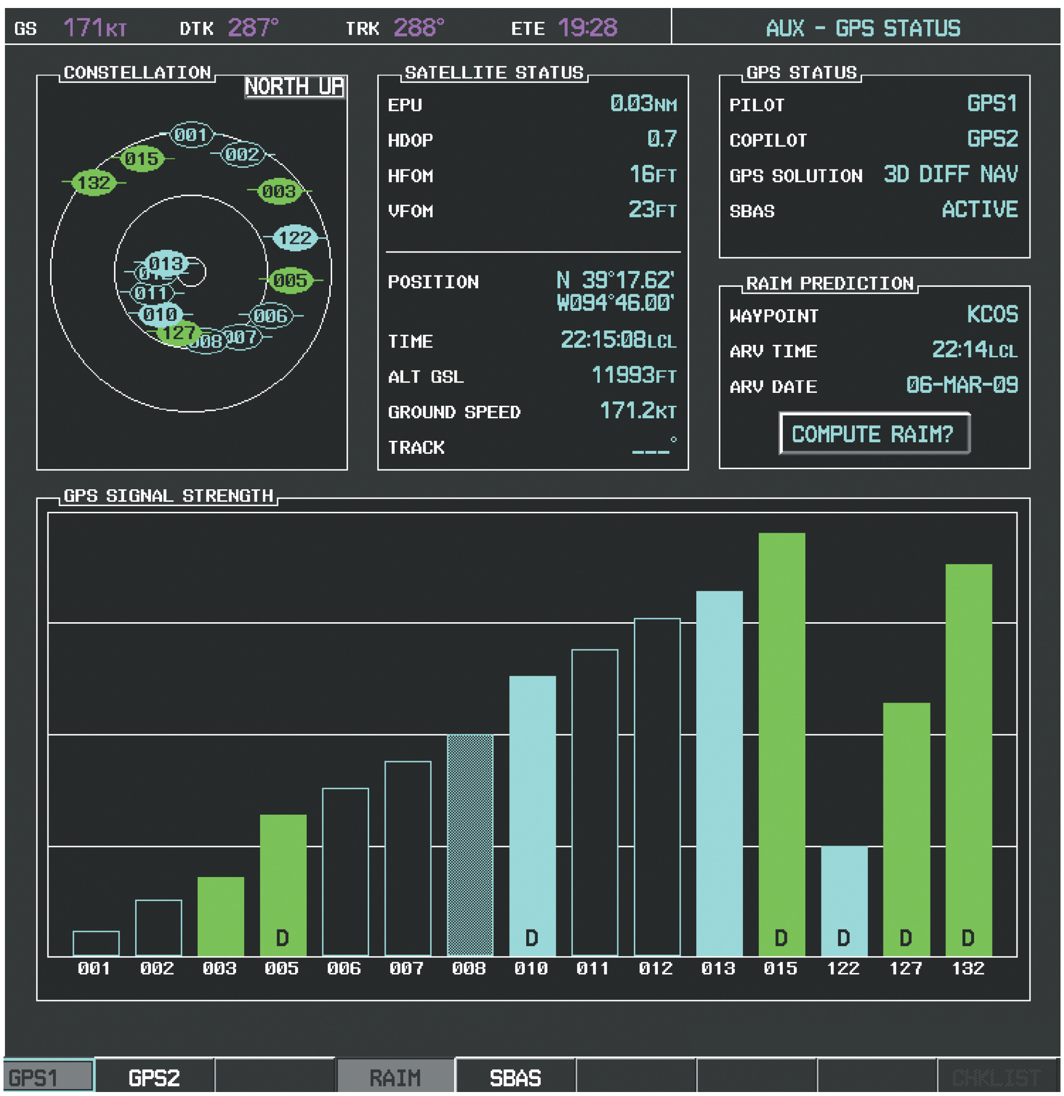
(g) Close PFD circuit breaker and de-activate pilot-side reversion mode. Verify PFD1 and MFD return to normal display modes.

Reversionary Mode
Figure 55

(11) GPS Signal Acquisition

See "Figure 56".

Select the GPS STATUS page on the MFD (4th page in AUX group). Toggle between GPS 1 and GPS 2 using the two softkeys on the bottom of the display. Verify that both receivers show 3D DIFF NAV next to GPS SOLUTION in the GPS STATUS field on the MFD. (The GIA 63W units should normally acquire a 3D GPS navigation solution within 2 minutes of startup, provided the aircraft is outside or indoors with a GPS repeater).



GPS Status Page
Figure 56

(12) Engine Run Checks

The following verify certain annunciations and displays.

WARNING: DO NOT WORK IN THE ENGINE AREA WHILE THE ENGINE IS RUNNING.

CAUTION: AFTER RUNNING THE ENGINE, THE EXHAUST STACKS WILL BE HOT.

(a) With engine OFF and PFD1 in reversionary mode.

Verify the difference between indicated left and right fuel quantity is LESS THAN 5 gallons.

(b) Remove fuses F411 and F412. (Simulates alternator failure.)

(c) START engine.

(d) Verify Alternator Operation:

(e) Verify the following annunciations with engine ON:

Turn ON Alternator. Verify ALTERNATOR FAIL annunciation.

(f) Turn OFF Engine

(g) Reinstall fuses F411 and F412.

(h) Restart engine.

(i) Verify Hour Meter operation

(j) Verify MAP, RPM, TIT, Fuel Flow, CHT, Oil Pressure, and Oil Temp indications are operating in normal range.

(k) Verify alternator #1 and alternator #2 current indications.

(l) With the Battery Master Switch ON and Alternator #1 and Alternator #2 on line, turn the Battery Master and Alternator #1 OFF.

1) Record Alternator # 2 current.

2) Turn the Alternator #1 ON and alternator # 2 off.

3) Record Alternator #1 current.

4) Compare Alternator #1 and #2 current readings and verify they are equal (within 3 amps).

(13) Flight Test

A flight test is recommended after the installation is complete to ensure satisfactory performance of the G1000.

(14) VHF Comm Tests

Verify the communications capability on both the high and low ends of the VHF COM spectrum.

(15) VOR/ILS Tests

Select a VOR channel within a 40 nautical mile range. Listen to the VOR audio and verify that no electrical interference such as magneto noise is present. Check the tone identifier filter operation. Fly inbound or outbound on a selected VOR radial and check for proper LEFT / RIGHT, TO / FROM flag indications on the CDI. Check the VOR accuracy.

(16) Maintenance Records

After conducting the function check flight, the airplane may be returned to service. Record the following information in the appropriate airplane maintenance logs:

(a) Part number of the loader card used to perform software loading or software updates.

(b) Any other applicable information related to the maintenance work performed on the airplane.

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CHART 38
G1000 - RETURN TO SERVICE CHECK LIST

Group System	SW Ver. OK	System	SW Ver. OK	System	SW Ver. OK
System Software					
PA-46 Image File	______				
Control & Display:					
PFD1	______	PFD2 FPGA	______	GCU	______
PFD1 FPGA	______	MFD1	______	GMC	______
PFD2	______	MFD1 FPGA	______		
CNS:					
GIA1	______	GTX1	______	COM1 (2)	______
GIA2	______	GTX2	______	COM2 (2)	______
GMA1	______	GIA1 AUDIO	______	NAV1 (2)	______
GMA2	______	GIA2 AUDIO	______	NAV1 FPGA (2-3)	______
GPS1	______	GDL69 (1)	______	NAV2 (2)	______
GPS2	______	GWX	______	NAV2 FPGA (2-3)	______
GS1 (2)	______	GWX FPGA	______		
GS2 (2)	______				
Sensors:					
GDC1	______	GMU1	______	GRS1	______
GDC1 FPGA	______	GMU1 FPGA	______	GRS1 FPGA	______
GDC2	______	GMU2	______	GRS2	______
GDC2 FPGA	______	GMU2 FPGA	______	GRS2 FPGA	______
GEA1	______				
AFCS:					
GSA PTCH	______	GSA ROLL	______		
GSA PTCH TRM	______	GSA YAW	______		
AFCS Cert Gains:					
GFC CERT GIA1	______	GFC CERT PT C	______	GFC CERT YC	______
GFC CERT GIA2	______	GFC CERT PT M	______	GFC CERT YM	______
GFC CERT PC	______	GFC CERT RC	______		
GFC CERT PM	______	GFC CERT RM	______		

(1) If installed.
(2) With S/W Version 0720.13 and up.
(3) Only if "-20" GIA installed.

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.11 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
GCU	006-B0472-20	3.01
GDC1	006-B0261-15	3.05
GDC1 FPGA	006-C0055-00	01.05
GDC2	006-B0261-15	3.05
GDC2 FPGA	006-C0055-00	01.05
GDL69	006-B0317-16	3.40.00 (Verify if installed)
GEA1	006-B0193-05	2.07
GFC CERT GIA1	006-D2018-00	2.00
GFC CERT GIA2	006-D2018-00	2.00
GFC CERT P C	006-D2018-00	2.00
GFC CERT P M	006-D2018-00	2.00
GFC CERT PT C	006-D2018-00	2.00
GFC CERT PT M	006-D2018-00	2.00
GFC CERT R C	006-D2018-00	2.00
GFC CERT R M	006-D2018-00	2.00
GFC CERT Y C	006-D2018-00	2.00
GFC CERT Y M	006-D2018-00	2.00
GIA1	006-B0544-32	6.05
GIA1 AUDIO	006-D0425-06	2.11
GIA2	006-B0544-32	6.05
GIA2 AUDIO	006-D0425-06	2.11
GMA1	006-B0203-43	4.03
GMC	006-B0387-20	3.00
GMU1	006-B0224-00	2.01
GMU1 FPGA	006-C0048-00	2.00
GMU2	006-B0224-00	2.01
GMU2 FPGA	006-C0048-00	2.00
GPS1	006-B0339-0A	3.2
GPS2	006-B0339-0A	3.2
GRS1	006-B0223-0A	2.12
GRS1 FPGA	006-C0049-00	02.00

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.11 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
GRS2	006-B0223-0A	2.12
GRS2 FPGA	006-C0049-00	02.00
GSA PTCH	006-B0398-32	2.30
GSA PTCH TRM	006-B0398-32	2.30
GSA ROLL	006-B0398-32	2.30
GSA YAW	006-B0398-32	2.30
GTX1	006-B0172-XX	6.11
GTX2	006-B0172-XX	6.11 (Verify if installed)
GWX	006-B0266-10	2.11 (Verify if installed)
GWX FPGA	006-C0042-06	3.01 (Verify if installed)
MFD1	006-B0319-81	10.01
MFD1 FPGA	006-C0036-04	1.04
PFD1	006-B0319-81	10.01
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-81	10.01
PFD2 FPGA	006-C0036-04	1.04
PA-46 IMAGE	006-B0720-11	0720.11 (1-2)

(1) PIR-PPS55026, Rev. B.
(2) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.12 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
GCU	006-B0472-20	3.01
GDC1	006-B0261-15	3.05
GDC1 FPGA	006-C0055-00	01.05
GDC2	006-B0261-15	3.05
GDC2 FPGA	006-C0055-00	01.05
GDL69	006-B0317-16	3.40.00 (Verify if installed)
GEA1	006-B0193-05	2.07
GFC CERT GIA1	006-D2018-00	2.00
GFC CERT GIA2	006-D2018-00	2.00
GFC CERT P C	006-D2018-00	2.00
GFC CERT P M	006-D2018-00	2.00
GFC CERT PT C	006-D2018-00	2.00
GFC CERT PT M	006-D2018-00	2.00
GFC CERT R C	006-D2018-00	2.00
GFC CERT R M	006-D2018-00	2.00
GFC CERT Y C	006-D2018-00	2.00
GFC CERT Y M	006-D2018-00	2.00
GIA1	006-B0544-34	6.07
GIA1 AUDIO	006-D0425-06	2.11
GIA2	006-B0544-34	6.07
GIA2 AUDIO	006-D0425-06	2.11
GMA1	006-B0203-43	4.03
GMC	006-B0387-20	3.00
GMU1	006-B0224-00	2.01
GMU1 FPGA	006-C0048-00	2.00
GMU2	006-B0224-00	2.01
GMU2 FPGA	006-C0048-00	2.00
GPS1	006-B0339-0A	3.2
GPS2	006-B0339-0A	3.2
GRS1	006-B0223-0A	2.12
GRS1 FPGA	006-C0049-00	02.00

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.12 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
GRS2	006-B0223-0A	2.12
GRS2 FPGA	006-C0049-00	02.00
GSA PTCH	006-B0398-32	2.30
GSA PTCH TRM	006-B0398-32	2.30
GSA ROLL	006-B0398-32	2.30
GSA YAW	006-B0398-32	2.30
GTX1	006-B0172-XX	6.11
GTX2	006-B0172-XX	6.11 (Verify if installed)
GWX	006-B0266-10	2.11 (Verify if installed)
GWX FPGA	006-C0042-06	3.01 (Verify if installed)
MFD1	006-B0319-81	10.01
MFD1 FPGA	006-C0036-04	1.04
PFD1	006-B0319-81	10.01
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-81	10.01
PFD2 FPGA	006-C0036-04	1.04
PA-46 IMAGE	006-B0720-12	0720.12 (1-2)

(1) PIR-PPS55026, Rev. D.
(2) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.13 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM1	006-B0081-XX	7.00	Only when (-10) GIA installed
COM2	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM2	006-B0081-XX	7.00	Only when (-10) GIA installed
GCU	006-B0472-20	3.01	
GDC1	006-B0261-18	3.08	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-18	3.08	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-16	3.40.00 (Verify if installed)	
GEA1	006-B0193-05	2.07	
GFC CERT GIA1	006-D2018-00	2.00	
GFC CERT GIA2	006-D2018-00	2.00	
GFC CERT P C	006-D2018-00	2.00	
GFC CERT P M	006-D2018-00	2.00	
GFC CERT PT C	006-D2018-00	2.00	
GFC CERT PT M	006-D2018-00	2.00	
GFC CERT R C	006-D2018-00	2.00	
GFC CERT R M	006-D2018-00	2.00	
GFC CERT Y C	006-D2018-00	2.00	
GFC CERT Y M	006-D2018-00	2.00	
GIA1	006-B0544-36	6.20	
GIA1 AUDIO	006-D0425-08	2.13	
GIA2	006-B0544-36	6.20	
GIA2 AUDIO	006-D0425-08	2.13	
GMA1	006-B0203-43	4.03	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-00	2.01	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-00	2.01	
GMU2 FPGA	006-C0048-00	2.00	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.13 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GPS1	006-B0339-0A	3.2	
GPS2	006-B0339-0A	3.2	
GRS1	006-B0223-22	3.02	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-22	3.02	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0083-XX	4.00	Only when (-10) GIA installed
GS2	006-B0083-XX	4.00	Only when (-10) GIA installed
GSA PTCH	006-B0398-32	2.30	
GSA PTCH TRM	006-B0398-32	2.30	
GSA ROLL	006-B0398-32	2.30	
GSA YAW	006-B0398-32	2.30	
GTX1	006-B0172-XX	6.11	
GTX2	006-B0172-XX	6.11 (Verify if installed)	
GWX	006-B0266-10	2.11 (Verify if installed)	
GWX FPGA	006-C0042-06	3.01 (Verify if installed)	
MFD1	006-B0319-A1	12.01	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV1 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV1	006-B0082-XX	5.03	Only when (-10) GIA installed
NAV2	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV2 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV2	006-B0082-XX	5.03	Only when (-10) GIA installed
PFD1	006-B0319-A1	12.01	
PFD1 FPGA	006-C0036-04	1.04	
PFD2	006-B0319-A1	12.01	
PFD2 FPGA	006-C0036-04	1.04	
PA-46 IMAGE	006-B0720-13	0720.13 (2-3)	

(1) Per Garmin ECM 11-1010(3), the COM and NAV software p/n may be displayed as "-XX" rather than the specific dash number. This is normal, the first 8 digits are sufficient to uniquely identify the software.
(2) PIR-PPS55026, Rev. G.
(3) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.14 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM1	006-B0081-XX	7.00	Only when (-10) GIA installed
COM2	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM2	006-B0081-XX	7.00	Only when (-10) GIA installed
GCU	006-B0472-20	3.01	
GDC1	006-B0261-18	3.08	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-18	3.08	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-16	3.40.00 (Verify if installed)	
GEA1	006-B0193-05	2.07	
GFC CERT GIA1	006-D2018-00	2.00	
GFC CERT GIA2	006-D2018-00	2.00	
GFC CERT P C	006-D2018-00	2.00	
GFC CERT P M	006-D2018-00	2.00	
GFC CERT PT C	006-D2018-00	2.00	
GFC CERT PT M	006-D2018-00	2.00	
GFC CERT R C	006-D2018-00	2.00	
GFC CERT R M	006-D2018-00	2.00	
GFC CERT Y C	006-D2018-00	2.00	
GFC CERT Y M	006-D2018-00	2.00	
GIA1	006-B0544-36	6.20	
GIA1 AUDIO	006-D0425-08	2.13	
GIA2	006-B0544-36	6.20	
GIA2 AUDIO	006-D0425-08	2.13	
GMA1	006-B0203-43	4.03	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-00	2.01	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-00	2.01	
GMU2 FPGA	006-C0048-00	2.00	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.14 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GPS1	006-B0339-0A	3.2	
GPS2	006-B0339-0A	3.2	
GRS1	006-B0223-22	3.02	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-22	3.02	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0083-XX	4.00	Only when (-10) GIA installed
GS2	006-B0083-XX	4.00	Only when (-10) GIA installed
GSA PTCH	006-B0398-32	2.30	
GSA PTCH TRM	006-B0398-32	2.30	
GSA ROLL	006-B0398-32	2.30	
GSA YAW	006-B0398-32	2.30	
GTX1	006-B0172-XX	6.11	
GTX2	006-B0172-XX	6.11 (Verify if installed)	
GWX	006-B0266-10	2.11 (Verify if installed)	
GWX FPGA	006-C0042-06	3.01 (Verify if installed)	
MFD1	006-B0319-A1	12.01	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV1 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV1	006-B0082-XX	5.03	Only when (-10) GIA installed
NAV2	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV2 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV2	006-B0082-XX	5.03	Only when (-10) GIA installed
PFD1	006-B0319-A1	12.01	
PFD1 FPGA	006-C0036-04	1.04	
PFD2	006-B0319-A1	12.01	
PFD2 FPGA	006-C0036-04	1.04	
PA-46 IMAGE	006-B0720-14	0720.14 (2-3)	

(1) Per Garmin ECM 11-1010(3), the COM and NAV software p/n may be displayed as “-XX” rather than the specific dash number. This is normal, the first 8 digits are sufficient to uniquely identify the software.
(2) PIR-PPS55026, Rev. H.
(3) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.15 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM1	006-B0081-XX	7.00	Only when (-10) GIA installed
COM2	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM2	006-B0081-XX	7.00	Only when (-10) GIA installed
GCU	006-B0472-20	3.01	
GDC1	006-B0261-18	3.08	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-18	3.08	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-16	3.40.00 (Verify if installed)	
GEA1	006-B0193-05	2.07	
GFC CERT GIA1	006-D2018-00	2.00	
GFC CERT GIA2	006-D2018-00	2.00	
GFC CERT P C	006-D2018-00	2.00	
GFC CERT P M	006-D2018-00	2.00	
GFC CERT PT C	006-D2018-00	2.00	
GFC CERT PT M	006-D2018-00	2.00	
GFC CERT R C	006-D2018-00	2.00	
GFC CERT R M	006-D2018-00	2.00	
GFC CERT Y C	006-D2018-00	2.00	
GFC CERT Y M	006-D2018-00	2.00	
GIA1	006-B0544-36	6.40	
GIA1 AUDIO	006-D0425-08	2.13	
GIA2	006-B0544-36	6.40	
GIA2 AUDIO	006-D0425-08	2.13	
GMA1	006-B0203-43	4.03	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-00	2.01	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-00	2.01	
GMU2 FPGA	006-C0048-00	2.00	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.15 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GPS1	006-B0339-0A	3.2	
GPS2	006-B0339-0A	3.2	
GRS1	006-B0223-22	3.02	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-22	3.02	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0083-XX	4.00	Only when (-10) GIA installed
GS2	006-B0083-XX	4.00	Only when (-10) GIA installed
GSA PTCH	006-B0398-32	2.30	
GSA PTCH TRM	006-B0398-32	2.30	
GSA ROLL	006-B0398-32	2.30	
GSA YAW	006-B0398-32	2.30	
GTX1	006-B0172-XX	6.11	
GTX2	006-B0172-XX	6.11 (Verify if installed)	
GWX	006-B0266-10	2.11 (Verify if installed)	
GWX FPGA	006-C0042-06	3.01 (Verify if installed)	
MFD1	006-B0319-A1	12.01	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV1 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV1	006-B0082-XX	5.03	Only when (-10) GIA installed
NAV2	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV2 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV2	006-B0082-XX	5.03	Only when (-10) GIA installed
PFD1	006-B0319-A1	12.01	
PFD1 FPGA	006-C0036-04	1.04	
PFD2	006-B0319-A1	12.01	
PFD2 FPGA	006-C0036-04	1.04	
PA-46 IMAGE	006-B0720-15	0720.15 (2-3)	

(1) Per Garmin ECM 11-1010(3), the COM and NAV software p/n may be displayed as "-XX" rather than the specific dash number. This is normal, the first 8 digits are sufficient to uniquely identify the software.
(2) PIR-PPS55026, Rev. J.
(3) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.16 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM1	006-B0081-XX	7.00	Only when (-10) GIA installed
COM2	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM2	006-B0081-XX	7.00	Only when (-10) GIA installed
GCU	006-B0472-20	3.01	
GDC1	006-B0261-18	3.08	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-18	3.08	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-16	3.40.00 (Verify if installed)	
GEA1	006-B0193-05	2.07	
GFC CERT GIA1	006-D2018-00	2.00	
GFC CERT GIA2	006-D2018-00	2.00	
GFC CERT P C	006-D2018-00	2.00	
GFC CERT P M	006-D2018-00	2.00	
GFC CERT PT C	006-D2018-00	2.00	
GFC CERT PT M	006-D2018-00	2.00	
GFC CERT R C	006-D2018-00	2.00	
GFC CERT R M	006-D2018-00	2.00	
GFC CERT Y C	006-D2018-00	2.00	
GFC CERT Y M	006-D2018-00	2.00	
GIA1	006-B0544-36	6.40	
GIA1 AUDIO	006-D0425-08	2.13	
GIA2	006-B0544-36	6.40	
GIA2 AUDIO	006-D0425-08	2.13	
GMA1	006-B0203-43	4.03	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-00	2.01	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-00	2.01	
GMU2 FPGA	006-C0048-00	2.00	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.16 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GPS1	006-B0339-0A	3.2	
GPS2	006-B0339-0A	3.2	
GRS1	006-B0223-22	3.02	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-22	3.02	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0083-XX	4.00	Only when (-10) GIA installed
GS2	006-B0083-XX	4.00	Only when (-10) GIA installed
GSA PTCH	006-B0398-32	2.30	
GSA PTCH TRM	006-B0398-32	2.30	
GSA ROLL	006-B0398-32	2.30	
GSA YAW	006-B0398-32	2.30	
GTX1	006-B0172-XX	6.11	
GTX2	006-B0172-XX	6.11 (Verify if installed)	
GWX	006-B0266-12	2.20 (Verify if installed)	
GWX FPGA	006-C0042-06	3.01 (Verify if installed)	
MFD1	006-B0319-A1	12.01	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV1 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV1	006-B0082-XX	5.03	Only when (-10) GIA installed
NAV2	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV2 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV2	006-B0082-XX	5.03	Only when (-10) GIA installed
PFD1	006-B0319-A1	12.01	
PFD1 FPGA	006-C0036-04	1.04	
PFD2	006-B0319-A1	12.01	
PFD2 FPGA	006-C0036-04	1.04	
PA-46 IMAGE	006-B0720-16	0720.16 (2-3)	

(1) Per Garmin ECM 11-1010(3), the COM and NAV software p/n may be displayed as “-XX” rather than the specific dash number. This is normal, the first 8 digits are sufficient to uniquely identify the software.
(2) PIR-PPS55026, Rev. K.
(3) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.17 (Sheet 1 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM1	006-B0081-XX	7.00	Only when (-10) GIA installed
COM2	006-B0081-10	10.00 (1)	Only when (-20) GIA installed
COM2	006-B0081-XX	7.00	Only when (-10) GIA installed
GCU	006-B0472-20	3.01	
GDC1	006-B0261-18	3.08	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-18	3.08	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-16	4.02.00 (Verify if installed)	
GEA1	006-B0193-05	2.07	
GFC CERT GIA1	006-D2018-00	2.00	
GFC CERT GIA2	006-D2018-00	2.00	
GFC CERT P C	006-D2018-00	2.00	
GFC CERT P M	006-D2018-00	2.00	
GFC CERT PT C	006-D2018-00	2.00	
GFC CERT PT M	006-D2018-00	2.00	
GFC CERT R C	006-D2018-00	2.00	
GFC CERT R M	006-D2018-00	2.00	
GFC CERT Y C	006-D2018-00	2.00	
GFC CERT Y M	006-D2018-00	2.00	
GIA1	006-B0544-36	6.40	
GIA1 AUDIO	006-D0425-08	2.13	
GIA2	006-B0544-36	6.40	
GIA2 AUDIO	006-D0425-08	2.13	
GMA1	006-B0203-43	4.03	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-00	2.01	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-00	2.01	
GMU2 FPGA	006-C0048-00	2.00	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 720.17 (Sheet 2 of 2)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GPS1	006-B0339-0A	3.2	
GPS2	006-B0339-0A	3.2	
GRS1	006-B0223-22	3.02	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-22	3.02	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0083-XX	4.00	Only when (-10) GIA installed
GS2	006-B0083-XX	4.00	Only when (-10) GIA installed
GSA PTCH	006-B0398-32	2.30	
GSA PTCH TRM	006-B0398-32	2.30	
GSA ROLL	006-B0398-32	2.30	
GSA YAW	006-B0398-32	2.30	
GTX1	006-B0172-XX	6.11	
GTX2	006-B0172-XX	6.11 (Verify if installed)	
GWX	006-B0266-12	2.20 (Verify if installed)	
GWX FPGA	006-C0042-06	3.01 (Verify if installed)	
MFD1	006-B0319-A1	12.01	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV1 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV1	006-B0082-XX	5.03	Only when (-10) GIA installed
NAV2	006-B0082-11	6.01 (1)	Only when (-20) GIA installed
NAV2 FPGA	006-C0124-00	1.00 (1)	Only when (-20) GIA installed
NAV2	006-B0082-XX	5.03	Only when (-10) GIA installed
PFD1	006-B0319-A1	12.01	
PFD1 FPGA	006-C0036-04	1.04	
PFD2	006-B0319-A1	12.01	
PFD2 FPGA	006-C0036-04	1.04	
PA-46 IMAGE	006-B0720-17	0720.17 (2-3)	

(1) Per Garmin ECM 11-1010(3), the COM and NAV software p/n may be displayed as “-XX” rather than the specific dash number. This is normal, the first 8 digits are sufficient to uniquely identify the software.
(2) PIR-PPS55026, Rev. L.
(3) See "Figure 52".

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CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.02 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-XX	11.20	
COM2	006-B0081-XX	11.20	
GCU	006-B0472-20	3.01	
GDC1	006-B0261-20	3.10	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-20	3.10	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-22	4.02.00	Optional, Verify If Installed
GEA1	006-B0193-05	2.07	
GIA1	006-B0544-4M	7.70	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4M	7.70	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0773-08	3.08	
GMA1 BB	006-B0773-BB	2.00	
GMA1 RGN	006-D3035-10	2.00	
GMA1 ARC	006-D3035-35	3.05	
GMA1 AUX	006-B0772-05	2.22	
GMA1 AUX AUD	006-D3034-16	2.15	
GMA1 AUX BB	006-B0772-BB	2.00	
GMA1 AUX RGN	006-D3034-05	2.00	
GMA1 AUX ASR 1	006-D3034-14	2.20	
GMA1 AUX ASR 2	006-D3034-08	2.00	
GMA1 AUX ASR 3	006-D3034-09	2.00	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	
GMU2 FPGA	006-C0048-00	2.00	
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.02 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GRS1	006-B0223-24	3.04	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-24	3.04	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0082-XX	6.02	
GS2	006-B0082-XX	6.02	
GSA PTCH CTL	006-B0398-37	3.30	
GFC CERT P C	006-D2018-01	2.01	
GSA PTCH MON	006-B0398-37	3.30	
GFC CERT P M	006-D2018-01	2.01	
GSA PTCH TRM C	006-B0398-37	3.30	
GFC CERT PT C	006-D5863-00	2.00	
GSA PTCH TRM M	006-B0398-37	3.30	
GFC CERT PT M	006-D5863-00	2.00	
GSA ROLL CTL	006-B0398-37	3.30	
GFC CERT RC	006-D2018-01	2.01	
GSA ROLL MON	006-B0398-37	3.30	
GFC CERT R M	006-D2018-01	2.01	
GSA YAW CTL	006-B0398-37	3.30	
GFC CERT Y C	006-D2018-01	2.01	
GSA YAW MON	006-B0398-37	3.30	
GFC CERT Y M	006-D2018-01	2.01	
GTS	006-B1244-07	2.22	Optional, Verify If Installed
GTS FPGA	006-C0081-20	2.0	Optional, Verify If Installed
GTS RGN LIST	006-D2767-01	2.20	Optional, Verify If Installed
GTX1	006-B0172-XX	8.01	
GTX2	006-B0172-XX	8.01	Optional, Verify If Installed
GWX	006-B0266-11	2.12	Optional, Verify If Installed
GWX FPGA	006-C0042-06	3.01	Optional, Verify If Installed
MFD1	006-B0319-D0	15.00	
MFD1 FPGA	006-C0036-04	1.04	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.02 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
NAV1	006-B0082-12	6.02
NAV1 FPGA	006-C0124-00	1.0
NAV2	006-B0082-12	6.02
NAV2 FPGA	006-C0124-00	1.0
PFD1	006-B0319-D0	15.00
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-D0	15.00
PFD2 FPGA	006-C0036-04	1.04
PA-46-350P IMAGE	006-D2164-02	2164.02 (1-2)

(1) PIR-107976, Rev. B/F.
(2) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.03 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-XX	11.20	
COM2	006-B0081-XX	11.20	
GCU	006-B0472-20	3.01	
GDC1	006-B0261-20	3.10	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-20	3.10	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-22	4.02.00	Optional, Verify If Installed
GEA1	006-B0193-05	2.07	
GIA1	006-B0544-4M	7.70	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4M	7.70	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0773-08	3.08	
GMA1 BB	006-B0773-BB	2.00	
GMA1 RGN	006-D3035-10	2.00	
GMA1 ARC	006-D3035-35	3.05	
GMA1 AUX	006-B0772-05	2.22	
GMA1 AUX AUD	006-D3034-16	2.15	
GMA1 AUX BB	006-B0772-BB	2.00	
GMA1 AUX RGN	006-D3034-05	2.00	
GMA1 AUX ASR 1	006-D3034-14	2.20	
GMA1 AUX ASR 2	006-D3034-08	2.00	
GMA1 AUX ASR 3	006-D3034-09	2.00	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	
GMU2 FPGA	006-C0048-00	2.00	
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.03 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GRS1	006-B0223-24	3.04	
GRS1 FPGA	006-C0049-00	02.00	
GRS2	006-B0223-24	3.04	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0082-XX	6.02	
GS2	006-B0082-XX	6.02	
GSA PTCH CTL	006-B0398-37	3.30	
GFC CERT P C	006-D2018-01	2.01	
GSA PTCH MON	006-B0398-37	3.30	
GFC CERT P M	006-D2018-01	2.01	
GSA PTCH TRM C	006-B0398-37	3.30	
GFC CERT PT C	006-D5863-00	2.00	
GSA PTCH TRM M	006-B0398-37	3.30	
GFC CERT PT M	006-D5863-00	2.00	
GSA ROLL CTL	006-B0398-37	3.30	
GFC CERT RC	006-D2018-01	2.01	
GSA ROLL MON	006-B0398-37	3.30	
GFC CERT R M	006-D2018-01	2.01	
GSA YAW CTL	006-B0398-37	3.30	
GFC CERT Y C	006-D2018-01	2.01	
GSA YAW MON	006-B0398-37	3.30	
GFC CERT Y M	006-D2018-01	2.01	
GTS	006-B1244-07	2.22	Optional, Verify If Installed
GTS FPGA	006-C0081-20	2.0	Optional, Verify If Installed
GTS RGN LIST	006-D2767-01	2.20	Optional, Verify If Installed
GTX1	006-B0172-XX	8.01	
GTX2	006-B0172-XX	8.01	Optional, Verify If Installed
GWX	006-B0266-12	2.20	Optional, Verify If Installed
GWX FPGA	006-C0042-06	3.01	Optional, Verify If Installed
MFD1	006-B0319-D0	15.00	
MFD1 FPGA	006-C0036-04	1.04	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.03 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
NAV1	006-B0082-12	6.02
NAV1 FPGA	006-C0124-00	1.0
NAV2	006-B0082-12	6.02
NAV2 FPGA	006-C0124-00	1.0
PFD1	006-B0319-D0	15.00
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-D0	15.00
PFD2 FPGA	006-C0036-04	1.04
PA-46-350P IMAGE	006-D2164-03	2164.03 (1-2)

(1) PIR-107976, Rev. C/F.
(2) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.05 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-XX	11.20	
COM2	006-B0081-XX	11.20	
GCU	006-B0472-20	3.01	
GDC1	006-B0261-20	3.10	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-20	3.10	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-22	4.02.00	Optional, Verify If Installed
GEA1	006-B0193-05	2.07	
GIA1	006-B0544-4S	7.80	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4S	7.80	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0773-08	3.08	
GMA1 RGN	006-D3035-10	2.00	
GMA1 ARC	006-D3035-35	3.05	
GMA1 AUX	006-B0772-05	2.22	
GMA1 AUX AUD	006-D3034-16	2.15	
GMA1 AUX RGN	006-D3034-05	2.00	
GMA1 AUX ASR 1	006-D3034-14	2.20	
GMA1 AUX ASR 2	006-D3034-08	2.00	
GMA1 AUX ASR 3	006-D3034-09	2.00	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	
GMU2 FPGA	006-C0048-00	2.00	
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-24	3.04	
GRS1 FPGA	006-C0049-00	02.00	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.05 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GRS2	006-B0223-24	3.04	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0082-XX	6.02	
GS2	006-B0082-XX	6.02	
GSA PTCH CTL	006-B0398-37	3.30	
GFC CERT P C	006-D2018-01	2.01	
GSA PTCH MON	006-B0398-37	3.30	
GFC CERT P M	006-D2018-01	2.01	
GSA PTCH TRM C	006-B0398-37	3.30	
GFC CERT PT C	006-D5863-00	2.00	
GSA PTCH TRM M	006-B0398-37	3.30	
GFC CERT PT M	006-D5863-00	2.00	
GSA ROLL CTL	006-B0398-37	3.30	
GFC CERT RC	006-D2018-01	2.01	
GSA ROLL MON	006-B0398-37	3.30	
GFC CERT R M	006-D2018-01	2.01	
GSA YAW CTL	006-B0398-37	3.30	
GFC CERT Y C	006-D2018-01	2.01	
GSA YAW MON	006-B0398-37	3.30	
GFC CERT Y M	006-D2018-01	2.01	
GTS	006-B1244-50	3.00	Optional, Verify If Installed
GTS FPGA	006-C0081-20	2.0	Optional, Verify If Installed
GTS RGN LIST	006-D2767-02	3.00	Optional, Verify If Installed
GTX1	006-B0172-XX	8.02	
GTX2	006-B0172-XX	8.02	Optional, Verify If Installed
GWX	006-B0266-12	2.20	Optional, Verify If Installed
GWX FPGA	006-C0042-06	3.01	Optional, Verify If Installed
MFD1	006-B0319-D7	15.12	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-12	6.02	
NAV1 FPGA	006-C0124-00	1.0	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.05 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
NAV2	006-B0082-12	6.02
NAV2 FPGA	006-C0124-00	1.0
PFD1	006-B0319-D7	15.12
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-D7	15.12
PFD2 FPGA	006-C0036-04	1.04
PA-46-350P IMAGE	006-B2164-05	2164.05 (1-2)

(1) PIR-107976, Rev. D/F.
(2) See "Figure 52".

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.06 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B0081-XX	11.20	
COM2	006-B0081-XX	11.20	
GCU	006-B0472-20	3.01	
GDC1	006-B0261-20	3.10	
GDC1 FPGA	006-C0055-00	01.05	
GDC2	006-B0261-20	3.10	
GDC2 FPGA	006-C0055-00	01.05	
GDL69	006-B0317-22	4.02.00	Optional, Verify If Installed
GEA1	006-B0193-05	2.07	
GIA1	006-B0544-4S	7.80	
GIA1 AUDIO	006-D4432-00	2.00	
GIA2	006-B0544-4S	7.80	
GIA2 AUDIO	006-D4432-00	2.00	
GMA1	006-B0773-08	3.08	
GMA1 RGN	006-D3035-10	2.00	
GMA1 ARC	006-D3035-35	3.05	
GMA1 AUX	006-B0772-05	2.22	
GMA1 AUX AUD	006-D3034-16	2.15	
GMA1 AUX RGN	006-D3034-05	2.00	
GMA1 AUX ASR 1	006-D3034-14	2.20	
GMA1 AUX ASR 2	006-D3034-08	2.00	
GMA1 AUX ASR 3	006-D3034-09	2.00	
GMC	006-B0387-20	3.00	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	
GMU2 FPGA	006-C0048-00	2.00	
GPS1	006-B0339-20	5.0	
GPS2	006-B0339-20	5.0	
GRS1	006-B0223-24	3.04	
GRS1 FPGA	006-C0049-00	02.00	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.06 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GRS2	006-B0223-24	3.04	
GRS2 FPGA	006-C0049-00	02.00	
GS1	006-B0082-XX	6.02	
GS2	006-B0082-XX	6.02	
GSA PTCH CTL	006-B0398-37	3.30	
GFC CERT P C	006-D2018-01	2.01	
GSA PTCH MON	006-B0398-37	3.30	
GFC CERT P M	006-D2018-01	2.01	
GSA PTCH TRM C	006-B0398-37	3.30	
GFC CERT PT C	006-D5863-00	2.00	
GSA PTCH TRM M	006-B0398-37	3.30	
GFC CERT PT M	006-D5863-00	2.00	
GSA ROLL CTL	006-B0398-37	3.30	
GFC CERT RC	006-D2018-01	2.01	
GSA ROLL MON	006-B0398-37	3.30	
GFC CERT R M	006-D2018-01	2.01	
GSA YAW CTL	006-B0398-37	3.30	
GFC CERT Y C	006-D2018-01	2.01	
GSA YAW MON	006-B0398-37	3.30	
GFC CERT Y M	006-D2018-01	2.01	
GTS	006-B1244-50	3.00	Optional, Verify If Installed
GTS FPGA	006-C0081-20	2.0	Optional, Verify If Installed
GTS RGN LIST	006-D2767-02	3.00	Optional, Verify If Installed
GTX1	006-B0172-XX	8.02	
GTX2	006-B0172-XX	8.02	Optional, Verify If Installed
GWX	006-B0266-12	2.20	Optional, Verify If Installed
GWX FPGA	006-C0042-06	3.01	Optional, Verify If Installed
MFD1	006-B0319-D7	15.12	
MFD1 FPGA	006-C0036-04	1.04	
NAV1	006-B0082-12	6.02	
NAV1 FPGA	006-C0124-00	1.0	

CHART 39
REQUIRED EQUIPMENT LISTS (REL)

G1000 – VER. 2164.06 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
NAV2	006-B0082-12	6.02
NAV2 FPGA	006-C0124-00	1.0
PFD1	006-B0319-D7	15.12
PFD1 FPGA	006-C0036-04	1.04
PFD2	006-B0319-D7	15.12
PFD2 FPGA	006-C0036-04	1.04
PA-46-350P IMAGE	006-D2164-06	2164.06 (1-2)

(1) PIR-107976, Rev. E/F.
(2) See "Figure 52".

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INTEGRATED AVIONICS SYSTEM (IAS) - GARMIN G1000 NXi

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Integrated Avionics System (IAS) - Garmin G1000 NXi

(PIR-190-00907-00, Rev B.)

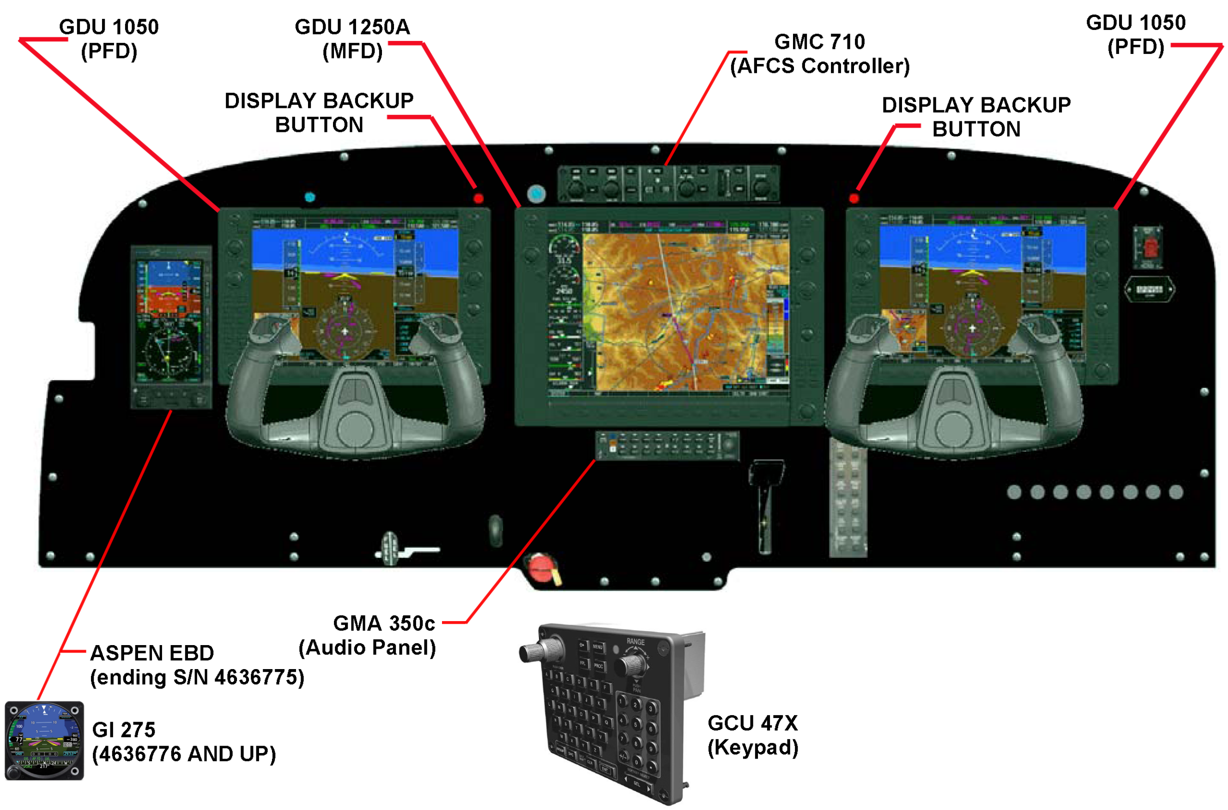
Standard equipment in 4636716, 4636720 and up.

A. Description

See "Figure 1".

The Garmin G1000 NXi Integrated Avionics System consists of three (3) displays, two (2) GDU 1050s dedicated as primary flight displays (PFD) for the pilot and copilot and a GDU 1250A multi-function display (MFD) installed in the center. Functions provided by the system include display of attitude, heading, navigation, traffic air data, engine and airframe status, and a moving map display with position derived by GPS. In addition to display functions, GPS navigation, VHF/Com, VOR/ILS and transponder functions are provided by the G1000 NXi system. Also the Garmin G1000 NXi system integrates the GFC 700 Automated Flight Control System and provides the flight director (FD), autopilot (AP), and yaw damper (YD) functions.

See the latest revision of the Garmin G1000 NXi Pilots Guide for Piper PA-46 M350 for control and operation information. See Vendor Publications under Supplementary Publications in the Introduction.



Garmin G1000 NXi Instrument Panel
Figure 1

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(1) Flight Instrumentation

See "Figure 2" on page 3425025.

The GRS 79 AHRSs, GDC 72 Air Data Computers, and GMU 44(B) Magnetometers are responsible for providing the G1000 NXi system with aircraft attitude, heading, altitude, airspeed, vertical speed, and outside air temperature information, all displayed on the PFD (data is displayed on the MFD in reversionary mode only).

Primary data outputs from the GRS and GDC are sent directly to the PFD via ARINC 429. Secondary data paths connect the GRS and GDC to the MFD. Additional communications paths connect the GRS and GDC to both GIA 64(W) units.

The GRS 79 receives GPS data from both GIAs, airspeed data from the GDC, and magnetic heading from the GMU. Using these three external sources, combined with internal sensor data, the GRS accurately calculates aircraft attitude and heading.

(2) Engine Indicator System

See "Figure 2" on page 3425025.

The GEA 71 provides engine/airframe data to the G1000 NXi system. Analog data is received from various sensors and transducers and is converted to a digital signal by the GEA 71B. Digital information is then sent through the primary RS-485 serial path to the #1 GIA. From the GIA, data is sent through the HSDB (ethernet) connection to the PFD, then on to the MFD for display. A backup data path from the GEA to the #2 GIA, then on to the MFD, exists in the event the primary path fails.

(3) Communications/Navigation Systems

See "Figure 2" on page 3425025.

The GIA 64(W) IAUs contain VHF COM, VHF NAV, and GPS receivers. COM and NAV audio is sent via digital audio to the GMA 350c Audio Panel.

GPS information is sent to the GRS 79 AHRS and to both displays for processing.

The GTX 33 Series Mode S and 3X5R Series Transponders communicates with both GIAs in a single transponder installation. In a dual installation, each transponder communicates with its on-side GIA. Transponder data is sent from the GIAs to the PFD where control and operation occurs.

(4) Components Details

See "Components" on page 34250267, for detailed descriptions of the individual G1000 NXi System Components / Line Replaceable Units (LRU’s).

See 22-10-00 for components of the GFC 700 Autopilot System.

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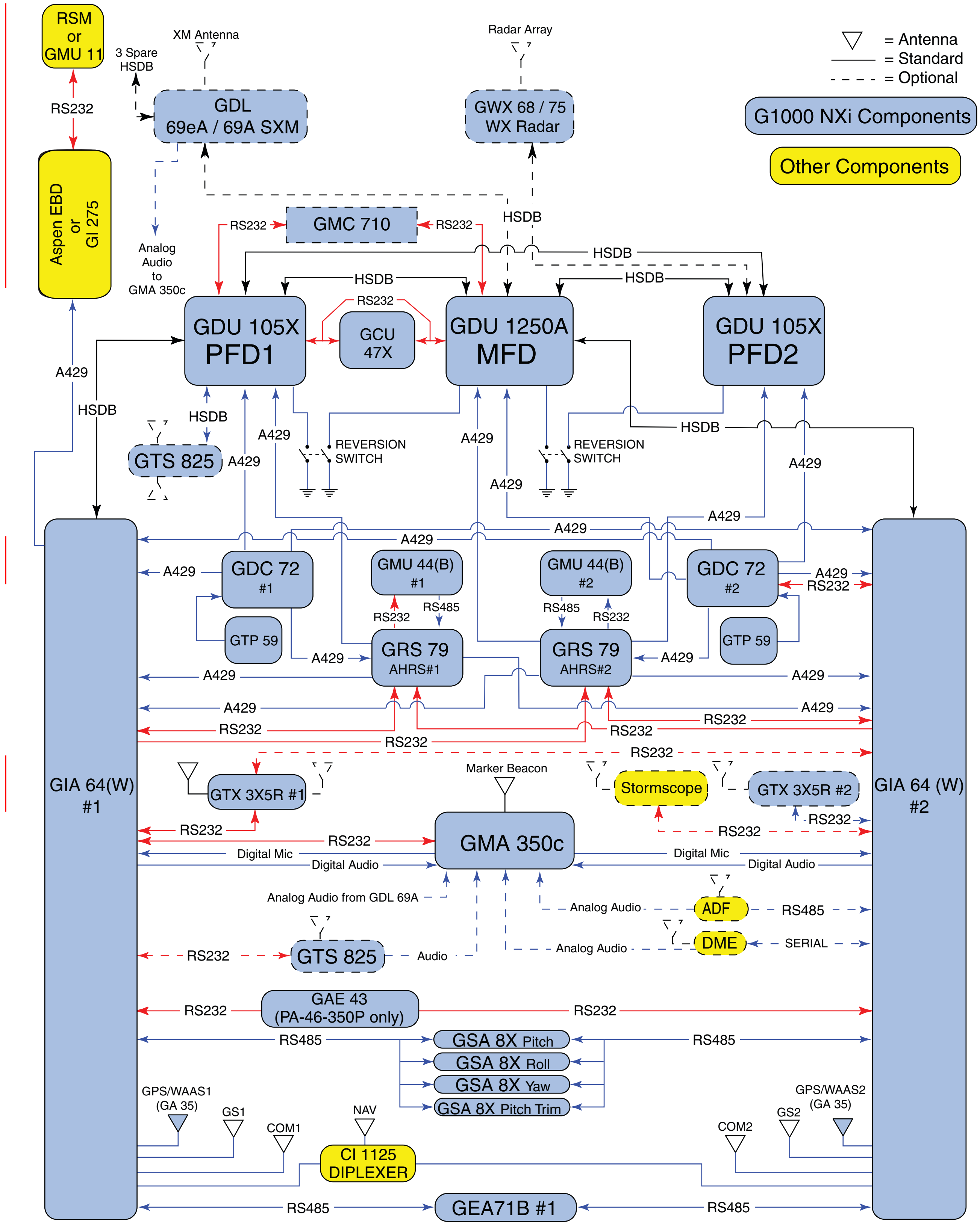


Component Block Diagram
Figure 2

B. Power Distribution

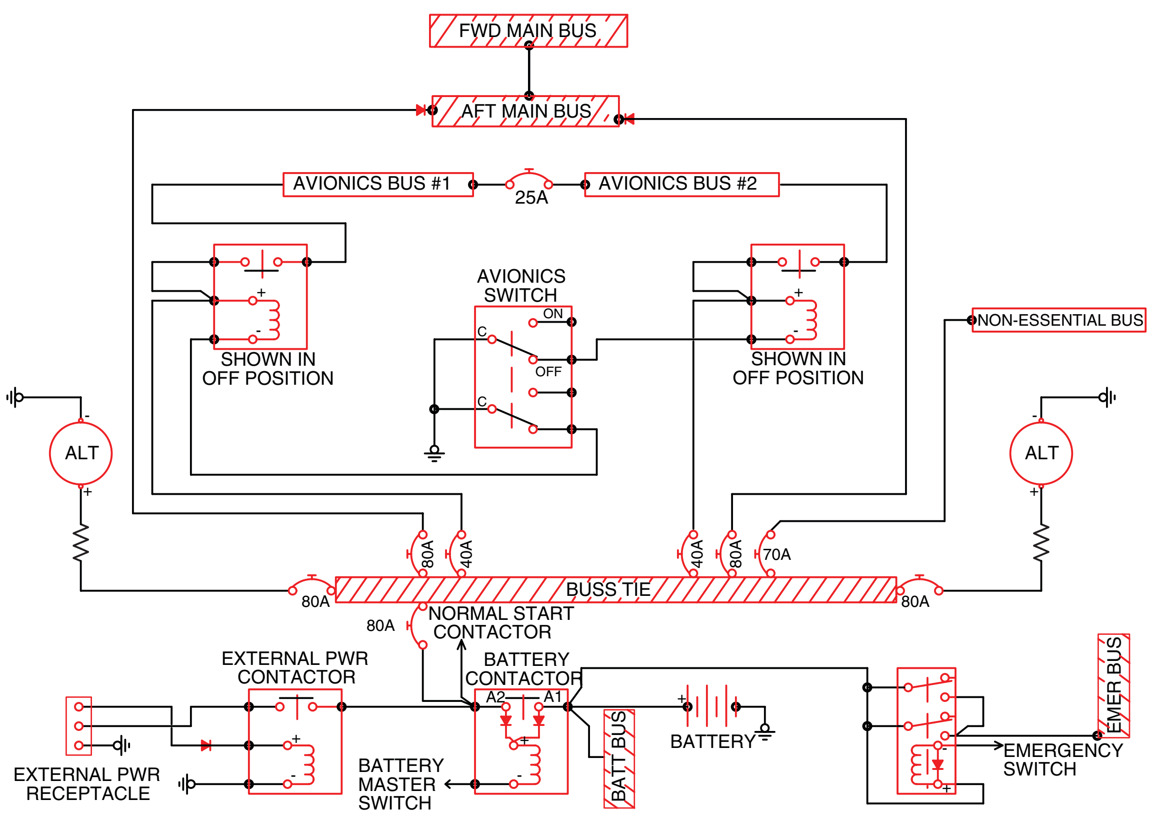
See "Figure 3" and "Figure 4".

Distribution of power to the G1000 NXi system occurs on four aircraft electrical busses.

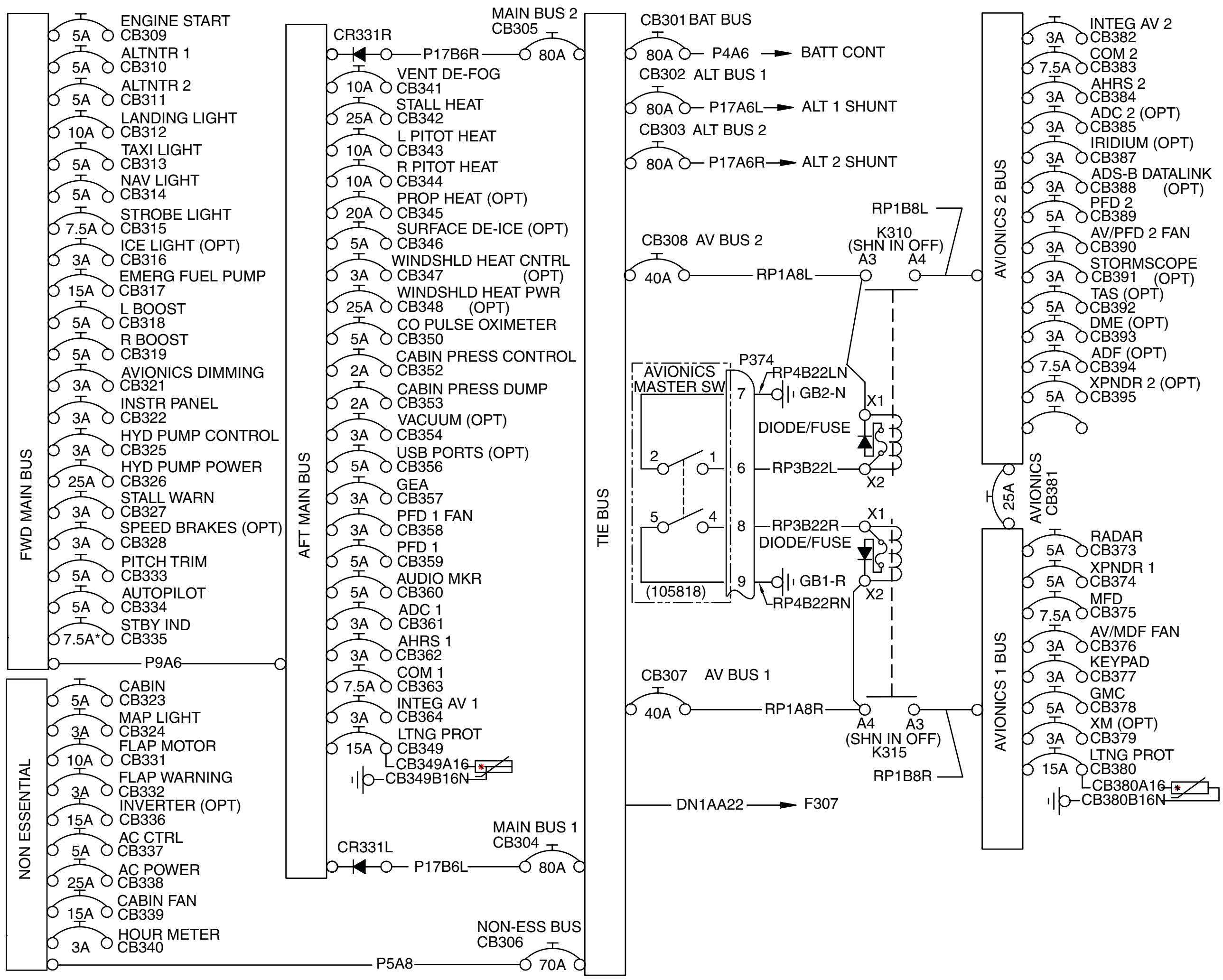
Main Bus: The current is fed to the main electrical bus by two conductors. Two in-line diodes provide isolation in the event of a ground fault in one of the feeder lines. The two feeders are protected by two 80 amp circuit breakers. The non-essential bus is fed by the 70 amp circuit breaker.

Emergency Bus: The Emergency bus exists as a backup power bus to the G1000 NXi system. When the EMER switch is turned on, power must be removed from the Main bus by turning the battery master switch to the “OFF” position. At the same time, the relay connecting the Emergency bus to the emergency battery is closed, providing emergency power to the connected equipment.

Avionics Bus: The two avionics busses are fed through independent contactors. The feeders to the contactors are protected by 40-amp circuit breakers. The busses are interconnected by a 25-amp bus tie and controlled by the avionics bus master switch. When the AV BUS MASTR switch on the overhead switch panel is pressed, both avionics contactors (normally closed) relax allowing current to flow to both avionics busses. Should the need arise, either avionics bus can be isolated by pulling the avionics bus BUS TIE circuit breaker and the appropriate avionics circuit breaker.



Bus Feeds
Figure 3



* 5A in 4636776 and up.

Component Power Sources
Figure 4

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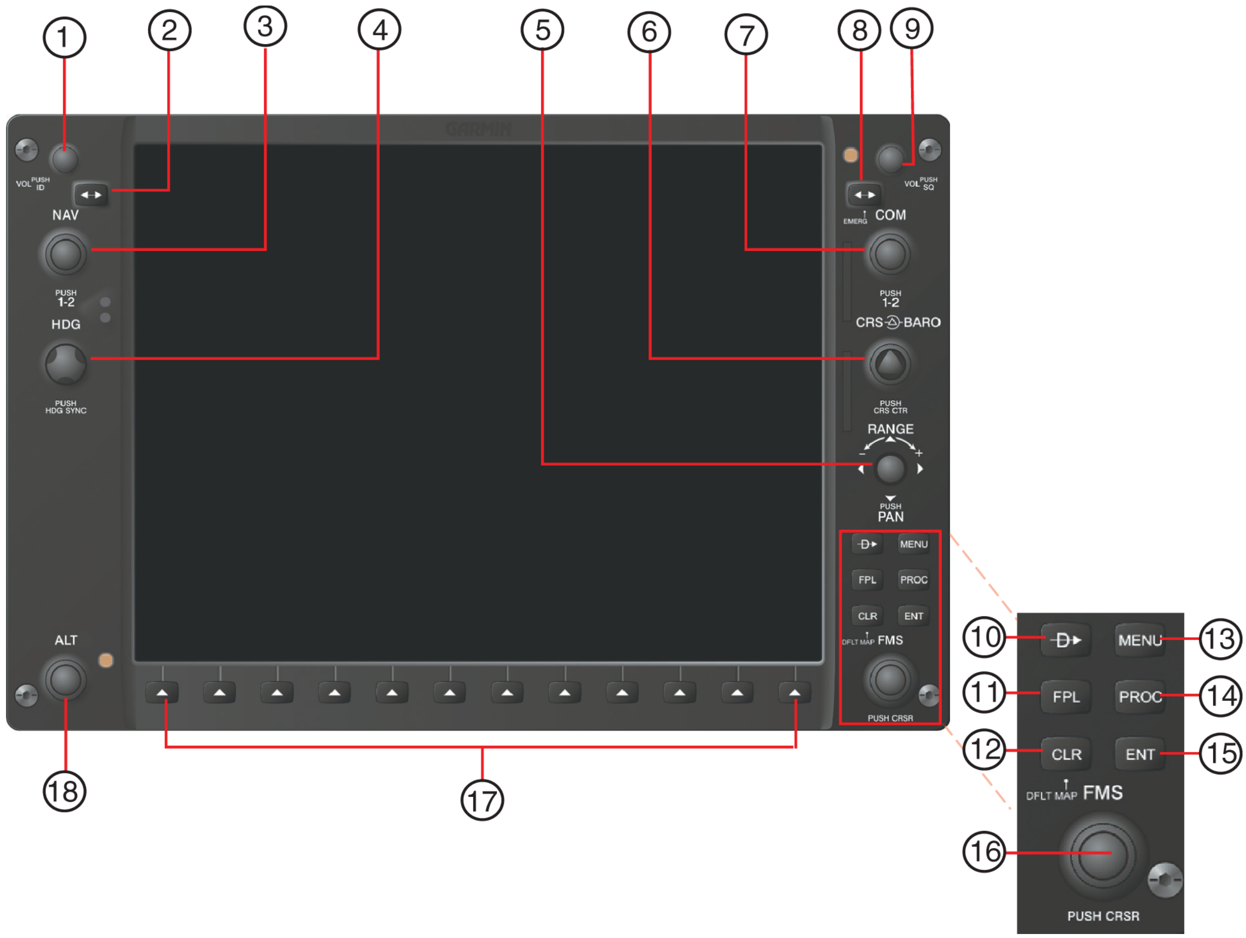
C. G1000 NXi System Controls and Operation

(1) System Controls

The G1000 NXi system controls are located on the PFD bezels, MFD/PFD Keypad Control Unit, AFCS Control Unit and above the respective PFD displays. See "Figure 5" on page 34250210 thru "Figure 13" on page 34250221 for a brief description.

The MFD/PFD Keypad Control Unit is a pedestal-mounted user interface allowing for ease of data entry, MFD/PFD operation, and NAV/COM tuning. Many procedures in this manual can be performed using the MFD/PFD Keypad Control Unit rather than the display bezel controls. Indicators above the PFD, MFD, NAV, and COM keys are illuminated when their respective control mode(s) are selected. The unit is in MFD control mode by default on system power-up. NAV/COM radio tuning can be accomplished in either PFD or MFD control mode. The appropriate frequency box on the selected display is outlined by a light blue selection box, which flashes for a few seconds to indicate Control Unit activity. Selection of a different display control or radio tuning mode results in cancelation of the previous radio tuning mode.

See the appropriate Garmin G1000 NXi Pilot's Guide for Piper PA-46 Mirage for detailed information on the operation and controls of the G1000 NXi system and its various components.



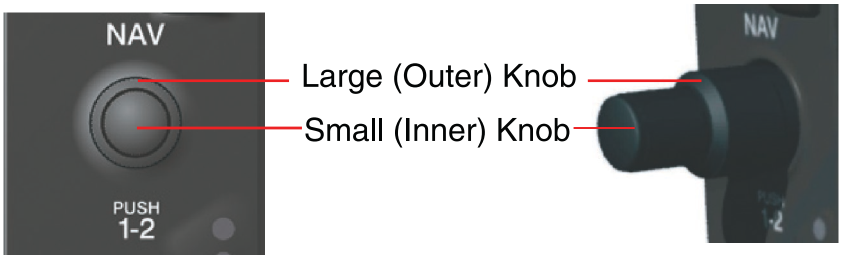
PFD CONTROLS

1	**NAV VOL/ID Knob**	Turn to control NAV audio volume (shown in the NAV Frequency Box as a percentage) Press to toggle Morse code identifier audio ON/OFF.
2	**NAV Frequency Transfer Key**	Transfers the standby and active NAV frequencies.
3	**NAV Knob**	Turn to tune NAV receiver standby frequencies (large knob for MHz; small for kHz) Press to toggle light blue tuning box between NAV1 and NAV2.
4	**Heading Knob**	Turn to manually select a heading Press to display a digital heading momentarily to the left of the HSI and synchronize the Selected Heading to the and current heading.
5	**Joystick**	Turn to change map range Press to activate Map Pointer for map panning.

PFD Controls
Figure 5 (Sheet 1 of 2)

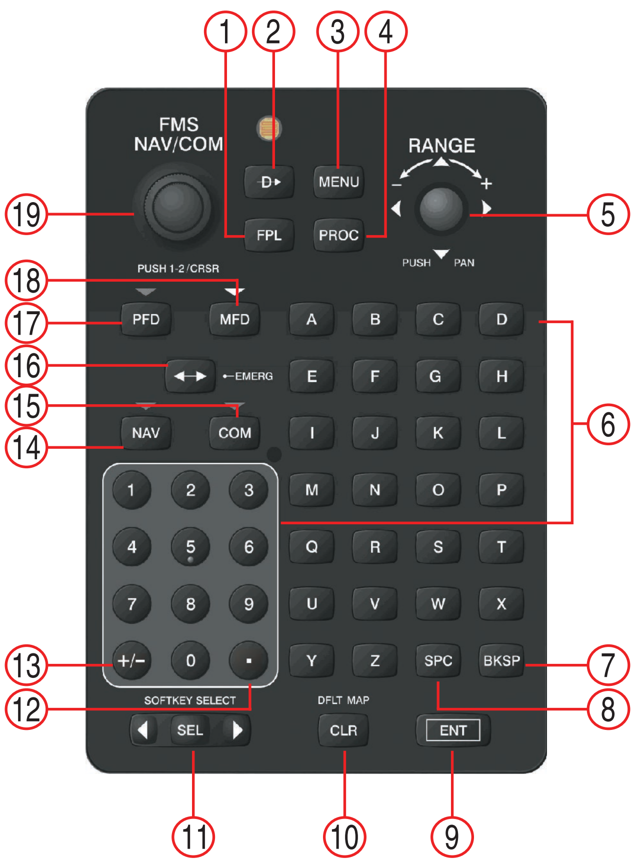
6 **CRS/BARO Knob**	Turn large knob for altimeter barometric pressure setting Turn small knob to adjust course (only when HSI is in VOR or OBS Mode) Press to re-center the CDI and return course pointer directly TO bearing of active waypoint/station.
7 **COM Knob**	Turn to tune COM transceiver standby frequencies (large knob for MHz; small for kHz) Press to toggle light blue tuning box between COM1 and COM2 The selected COM (green) is controlled with the COM MIC Key (Audio Panel).
8 **COM Frequency Transfer Key**	(EMERG) Transfers the standby and active COM frequencies Press and hold 2 seconds to tune the emergency frequency (121.5 MHz) automatically into the active frequency field.
9 **COM VOL/SQ Knob**	Turn to control COM audio volume level (shown as a percentage in the COM Frequency Box) Press to turn the COM automatic squelch ON/OFF.
10 **Direct-to Key** ( )	Activates the direct-to function and allows the user to enter a destination waypoint and establish a direct course to the selected destination (specified by identifier, chosen from the active route).
11 **FPL Key**	Displays flight plan information.
12 **CLR Key**	(DFLT MAP) Erases information, cancels entries, or removes menus Press and hold to display the MFD Navigation Map Page (MFD only).
13 **MENU Key**	Displays a context-sensitive list of options for accessing additional features or making setting changes.
14 **PROC Key**	Gives access to IFR departure procedures (DPs), arrival procedures (STARs), and approach procedures (IAPs) for a flight plan or selected airport.
15 **ENT Key**	Validates/confirms menu selection or data entry.
16 **FMS Knob** (Flight Management System Knob)	Press to turn the selection cursor ON/OFF. Data Entry: With cursor ON, turn to enter data in the highlighted field (large knob moves cursor location; small knob selects character for highlighted cursor location) Scrolling: When a list of information is too long for the window/box, a scroll bar appears, indicating more items to view. With cursor ON, turn large knob to scroll through the list. Page Selection: Turn knob on MFD to select the page to view (large knob selects a page group; small knob selects a specific page from the group).
17 **Softkey Selection Keys**	Press to select softkey shown above the bezel key on the PFD/MFD display.
18 **ALT Knob**	Sets the Selected Altitude, shown above the Altimeter (the large knob selects the thousands, the small knob selects the hundreds).

PFD Controls
Figure 5 (Sheet 2 of 2)



The NAV, CRS/BARO, COM, FMS, and ALT knobs are concentric dual knobs, each having small (inner) and large (outer) control portion. When a portion of the knob is not specified in the text, either may be used.

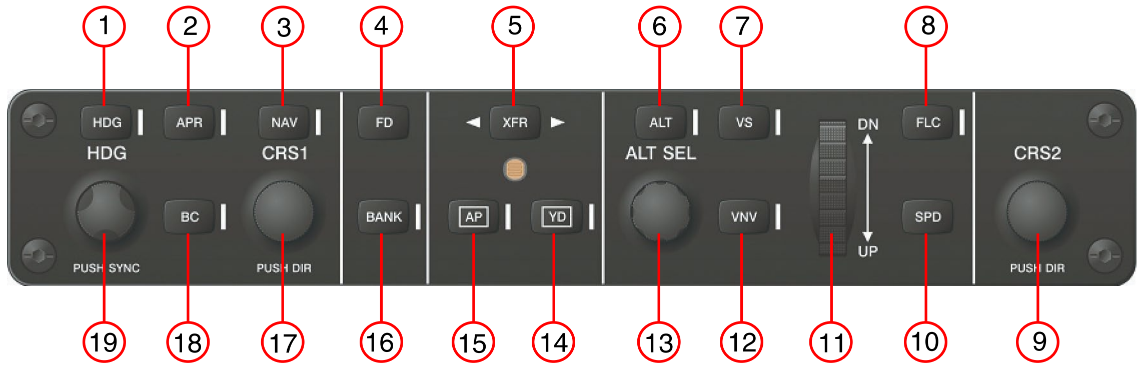
Concentric Dual Knobs
Figure 6



GCU 47X MFD/PFD Keypad Control Unit
Figure 7 (Sheet 1 of 2)

1	**FPL Key**	Displays flight plan information.
2	**Direct-to Key** ( )	Activates the direct-to function and allows the user to enter a destination waypoint and establish a direct course to the selected destination (specified by identifier, chosen from the active route).
3	**MENU Key**	Displays a context-sensitive list of options for accessing additional features or making setting changes.
4	**PROC Key**	Gives access to IFR departure procedures (DPs), arrival procedures (STARs), and approach procedures (IAPs) for a flight plan or selected airport.
5	**Joystick**	Turn to change map range Press to activate Map Pointer for map panning.
6	**Alphanumeric Keys**	Allow data entry (rather than using the FMS Knob to select characters/numbers).
7	**BKSP Key**	Moves cursor back one character space and removes last character entered.
8	**SPC Key**	Adds a space character.
9	**ENT Key**	Validates or confirms a menu selection or data entry.
10	**CLR Key**	Erases information, cancels entries, or removes menus. Press and hold to display the MFD Navigation Map Page (MFD only).
11	**SEL Key**	Arrows move light blue Softkey Selection Box on selected display Press the center to activate the selected softkey.
12	**Decimal Key**	Enters a decimal point character.
13	**Plus-Minus (±) Key**	Toggles entry between the + and - characters.
14	**NAV Key**	Selects/deselects NAV radio tuning mode on the MFD/PFD Control Unit.
15	**COM Key**	Selects/deselects COM radio tuning mode on the MFD/PFD Control Unit.
16	**Frequency Transfer Key**	(EMERG) Transfers between active and standby selected COM or NAV tuning frequencies Press and hold 2 seconds to tune the emergency frequency (121.5 MHz) automatically into the active frequency field.
17	**PFD Key**	When selected, the MFD/PFD Control Unit can be used to access PFD functions.
18	**MFD Key**	When selected, the MFD/PFD Control Unit can be used to access MFD functions (default display control mode).
19	**FMS/NAV-COM Knob**	NAV/COM Tuning Modes: Acts as the NAV or COM Knob PFD/MFD Control Modes: Acts as the FMS Knob.

GCU 476 MFD/PFD Keypad Control Unit
Figure 7 (Sheet 2 of 2)



The GFC 700 AFCS is mainly controlled through the GMC 710 AFCS Control Unit. The AFCS Control Unit consists of the following controls:

1 **HDG Key**	Selects/deselects Heading Select Mode.
2 **APR Key**	Selects/deselects Approach Mode.
3 **NAV Key**	Selects/deselects Navigation Mode.
4 **FD Key**	Activates/deactivates the flight director in the default pitch and roll modes. If the autopilot is engaged, the FD Key is disabled.
5 **XFR Key**	Switches the autopilot between the pilot-side and the copilot-side flight directors. This selection also selects which air data computer is communicating with the active transponder and which PFD triggers the altitude alert. Upon power-up, the pilot-side FD is selected.
6 **ALT Key**	Selects/deselects Altitude Hold Mode.
7 **VS Key**	Selects/deselects Vertical Speed Mode.
8 **FLC Key**	Selects/deselects Flight Level Change Mode.
9 **CRS2 Knob**	Sets the copilot-selected course on the HSI of PFD2 when the VOR1, VOR2, or OBS/SUSP mode is selected. Pressing this knob centers the CDI on the currently selected VOR. The copilot-selected course provides course reference to the copilot-side flight director when operating in Navigation and Approach modes.
10 **SPD Key**	Has no function.

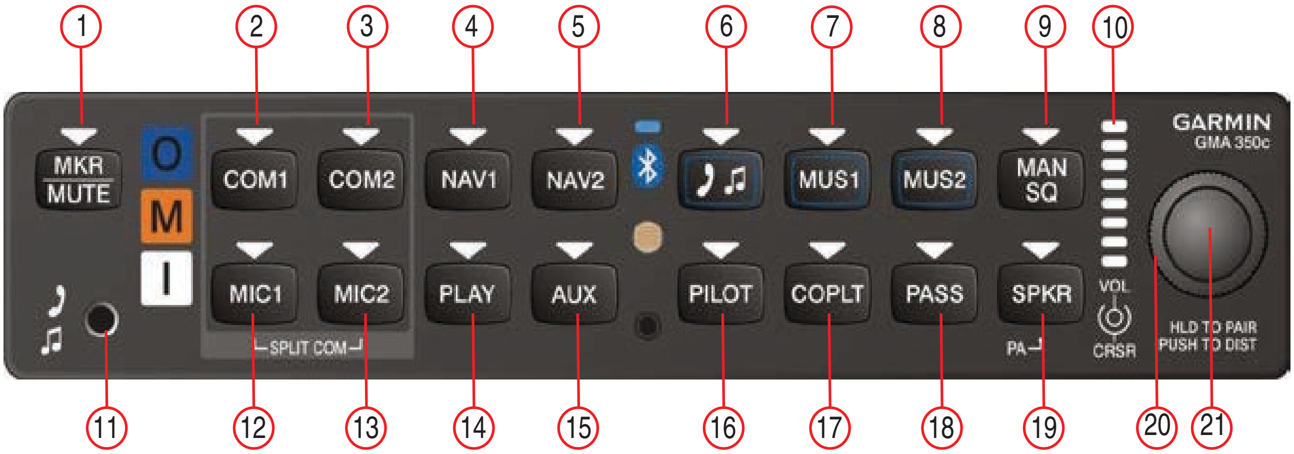
GMC 710 AFCS Control Unit
Figure 8 (Sheet 1 of 2)

11	**NOSE UP/DN Wheel**	Controls the active mode reference for the Pitch, Vertical Speed, and Flight Level Change modes.
12	**VNV Key**	Selects/deselects Vertical Navigation mode.
13	**ALT SEL Knob**	Sets the selected altitude in the Selected Altitude Box. In addition to providing the standard G1000 NXi altitude alerter function, selected altitude provides an altitude setting for the Altitude Capture/Hold mode of the AFCS.
14	**YD Key**	Engages/disengages the yaw damper.
15	**AP Key**	Engages/disengages the autopilot.
16	**BANK Key**	Selects/deselects Low Bank Mode.
17	**CRS1 Knob**	Sets the pilot-selected course on the HSI of PFD1 when the VOR1, VOR2, or OBS/SUSP mode is selected. Pressing this knob centers the CDI on the currently selected VOR. The pilot-selected course provides course reference to the pilot-side flight director when operating in Navigation and Approach modes.
18	**BC Key**	Selects/deselects Back Course Mode.
19	**HDG Knob**	Sets the selected heading on the HSI. When operating in Heading Select mode, this knob provides the heading reference to the flight director.

ADDITIONAL AFCS CONTROLS

The AP DISC (Autopilot Disconnect) Switch, CWS (Control Wheel Steering) Button, GO AROUND Switch, and MEPT (Manual Electric Pitch Trim) Switch are additional AFCS controls and are located in the cockpit, separately from the AFCS Control Unit.

GMC 710 AFCS Control Unit
Figure 8 (Sheet 2 of 2)



GMA 350c CONTROLS

NOTE: WHEN A KEY IS SELECTED, A TRIANGULAR ANNUNCIATOR ABOVE THE KEY IS ILLUMINATED.

1.	MKR/MUTE	Selects marker beacon receiver audio. Mutes the currently received marker beacon receiver audio. Deactivates automatically and marker beacon audio is heard when the next marker beacon signal is received. Also, stops play of recorded COM audio.
2.	COM1	When selected, audio from the #1 COM receiver can be heard. Press and hold to enable/disable monitored COM muting during primary COM reception.
3.	COM2	When selected, audio from the #2 COM receiver can be heard. Press and hold to enable/disable monitored COM muting during primary COM reception.
4.	NAV1	When selected, audio from the #1 NAV receiver can be heard.
5.	NAV2	When selected, audio from the #2 NAV receiver can be heard.
6.		Selects and deselects audio from a telephone or entertainment device connected to the Front Panel Jack. Audio from a telephone connected to the rear of the audio panel is used if a device is not connected to the Front Panel Jack. Press and hold to enable/disable  muting during reception.

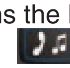
Audio Panel Controls (GMA 350c)
Figure 9 (Sheet 1 of 3)

7. MUS1	Selects and deselects music entertainment audio. Press and hold to enable/disable MUS1 muting during reception.
8. MUS2	Selects and deselects music entertainment audio. Press and hold to enable/disable MUS2 muting during reception.
9. MAN SQ	Manual Squelch annunciator. When lit, squelch is controlled manually.
10. Volume Indicator	Indicates volume/squelch setting relative to full scale.
11. Front Panel Jack	Used for an entertainment or telephone input.
12. MIC1	Selects the #1 transmitter for transmitting. COM1 receive is simultaneously selected when this key is pressed allowing received audio from the #1 COM receiver to be heard. COM2 receive can be added by pressing the COM2 Key. Selection of a second MIC button initiates Split-COM mode. When in Split-COM mode, the pilot is using COM1, the copilot is using the COM2.
13. MIC2	Selects the #2 transmitter for transmitting. COM2 receive is simultaneously selected when this key is pressed allowing received audio from the #2 COM receiver to be heard. COM1 receive can be added by pressing the COM1 Key. Selection of a second MIC button initiates Split-COM mode. When in Split-COM mode, the pilot is using COM1, the copilot is using the COM2.
14. PLAY	Press once to play the latest recorded memory block. Press while audio is playing begins playing the previously recorded memory block. Each subsequent press thereafter plays the previous block of memory.
15. AUX	When selected, audio from the AUX inputs can be heard.
16. PILOT	Controls the pilot intercom system. Press and hold to toggle 3D Audio on/off for all headset positions.
17. COPLT	Controls the copilot intercom system. Press and hold to toggle copilot configuration between crew and passenger.
18. PASS	Controls the passenger intercom system. Press and hold to enable/disable passenger muting during reception.
19. SPKR	Selects and deselects the cabin speaker. COM, NAV, AUX, and MKR receiver audio can be heard on the speaker. Press and hold for 2 seconds for Passenger Address (PA). The SPKR key flashes during PA.
20. Cursor (CRSR) Control Knob	Turn to move the cursor (flashing white or blue annunciator) to the desired source.
21. Volume (VOL) Control Knob	Turn the smaller knob to control volume or squelch of the selected source (indicated by the flashing white or blue annunciator). When the volume control cursor is not active press to switch to Blue-Select mode. If the volume control cursor is active, press twice (once to cancel the cursor, twice to activate Blue-Select mode).

Audio Panel Controls (GMA 350c)
Figure 9 (Sheet 2 of 3)

GMA 350c (BLUETOOTH®) CONTROLS

NOTE: WHEN A KEY IS SELECTED, A TRIANGULAR ANNUNCIATOR ABOVE THE KEY IS ILLUMINATED.

Bluetooth® Connection Annunciator	GMA 350c only - A flashing blue annunciator indicates the unit is discoverable. A solid blue annunciator indicates an active Bluetooth® connection.
 **Key Annunciator**	GMA 350c only - Assigns the Bluetooth® device to the  audio source. Press the  key until the annunciator turns blue. The key annunciator will cycle from OFF to WHITE to BLUE. WHITE selects the wired audio source and BLUE selects the Bluetooth® audio source.
22. MUS1 Key Annunciator	GMA 350c only - Assigns the Bluetooth® device to the MUS1 audio source. Press the MUS1 key until the annunciator turns blue. The key annunciator will cycle from OFF to WHITE to BLUE. WHITE selects the wired audio source and BLUE selects the Bluetooth® audio source.
23. MUS2 Key Annunciator	GMA 350c only - Assigns the Bluetooth® device to the MUS2 audio source. Press the MUS2 key until the annunciator turns blue. The key annunciator will cycle from OFF to WHITE to BLUE. WHITE selects the wired audio source and BLUE selects the Bluetooth® audio source.

NOTE: The Bluetooth® audio can only be assigned to one source at a time. Once the Bluetooth® audio is assigned to an audio source, the remaining entertainment audio sources will only cycle between OFF and WHITE.

24. Control Knob	GMA 350c only - Press and hold for two seconds to enable the GMA 350c as discoverable for pairing. The Bluetooth® Annunciator will flash to indicate that the unit is discoverable. The unit will remain discoverable for 90 seconds or until a successful pair is established. Once a successful pair is established, the audio "Bluetooth® paired" is played.

Audio Panel Controls (GMA 350c)
Figure 9 (Sheet 3 of 3)

(2) System Operation

The displays are connected together via multiple data busses, thus allowing for high-speed communication. As shown in "Figure 2" on page 3425025, each GIA 64 is connected to the on-side PFD. This section discusses the normal and reversionary modes of operation as well as the various AHRS modes of the G1000 NXi system.

(a) Normal Operation

1) PFD

In normal mode, the PFD presents graphical flight instrumentation (attitude, heading, airspeed, altitude and vertical speed), thereby replacing the traditional flight instrument cluster. The PFD also offers control for COM and NAV frequency selection.

2) MFD

In normal mode, the right portion of the MFD displays a full-color moving map with navigation information, while the left portion of the MFD is dedicated to the Engine Indication System (EIS). "Figure 10" gives an example of the G1000 NXi displays in normal mode.

Normal Mode
Figure 10

Reversionary Mode #1 PFD Fail
Figure 11

Reversionary Mode MFD Fail
Figure 12

(b) Reversionary Mode

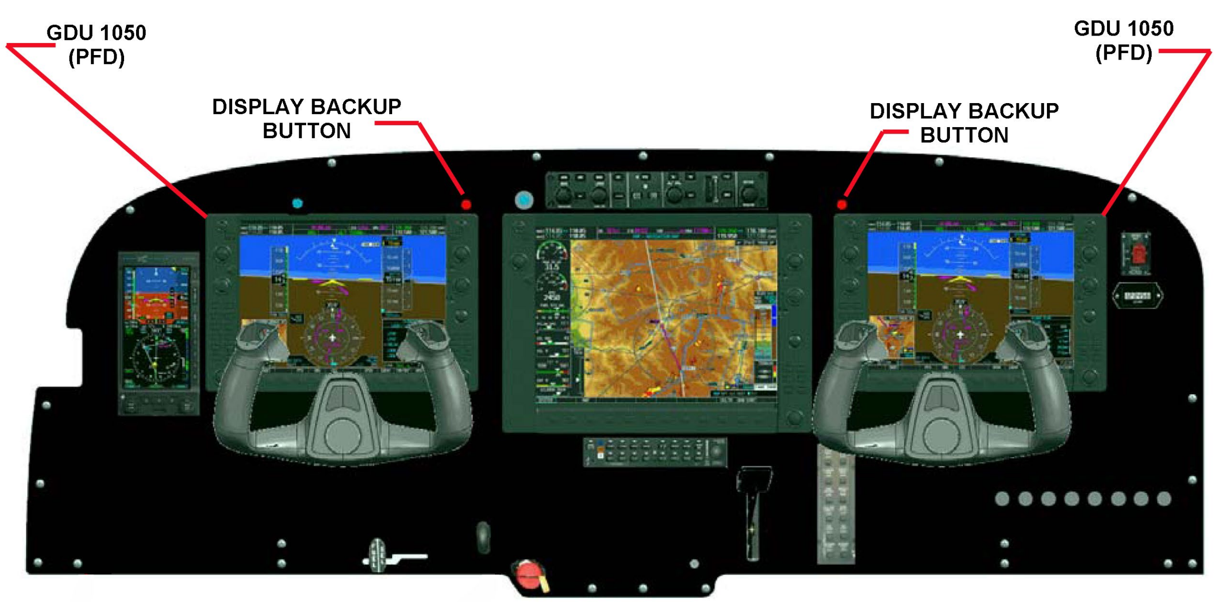
NOTE: The G1000 NXi system alerts the pilot when backup paths are utilized by the LRUs. Refer to the Appendices in the Garmin Pilot's Guide for further information regarding system-specific alerts.

Reversionary mode is a mode of operation in which all important flight information is presented identically on at least one of the remaining displays (see "Figure 11" and "Figure 12"). Transition to reversionary mode should be straight-forward for the pilot, flight parameters are presented in the same format as in normal mode. In the event of a PFD failure, the G1000 NXi system automatically switches to reversionary (backup) mode. In reversionary mode, all important flight information is presented on the remaining display(s) in the same format as in normal operating mode.

- PFD1 failure – MFD enters reversionary mode; PFD2 remains in normal mode.
- MFD failure – No automatic switch to reversionary mode. Either PFD may be put into reversionary mode by manually pressing the Display Backup Button above the respective PFD (see "Figure 13").
- PFD2 failure – MFD enters reversionary mode; PFD1 remains in normal mode. Reversionary mode can be activated manually by pressing the dedicated Display Backup Button above PFD1. Pressing this button again deactivates reversionary mode.

Each display can be configured to operate in reversionary mode, as follows:

- PFD1 – By pressing the Display Backup Button above the left PFD.
- MFD – By pressing the Display Backup Button above the left or right PFD.
- PFD2 – By pressing the Display Backup Button above the right PFD.



Display Backup Button
Figure 13

D. Troubleshooting

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, "Test" on page 341014, in 34-10-00.

Troubleshoot the G1000 NXi system by first identifying, then isolating the specific failure to the responsible LRU. There are several indications that the G1000 NXi presents to the pilot or technician, showing overall system condition. A course of action should be determined based on the information presented on the display.

The 5th AUX group page on the MFD (see "Figure 14") provides LRU health status by means of a green check or a red ‘X’. Also, LRU software versions are shown along with other database versions and dates. If a red ‘X’ is shown for an LRU, see the appropriate LRU troubleshooting chart for guidance.

Typically, the G1000 NXi Alerting System provides alerts/annunciations in conjunction with the information presented at the 5th AUX page.

CAUTION: "POST-INSTALLATION PROCEDURES" ON PAGE 342501171, BELOW, PROVIDE DETAILED INSTRUCTIONS ON EQUIPMENT CONFIGURATION AND RETURN-TO-SERVICE TESTING. ANYTIME A G1000 NXI COMPONENT OR LRU IS REMOVED, SWAPPED, OR REPLACED, THE TECHNICIAN MUST FOLLOW THE PROCEDURES GIVEN IN "POST-INSTALLATION PROCEDURES" ON PAGE 342501171 TO ENSURE PROPER OPERATION OF THE SYSTEM. SEE ALSO RETURN TO SERVICE UNDER THE INDIVIDUAL G1000 NXI COMPONENT IN "COMPONENTS" ON PAGE 34250162, BELOW.

Troubleshooting information is provided as follows:

(1) Line Replaceable Unit (LRU) Troubleshooting - see "Chart 5" on page 34250235.
(2) Engine / Airframe Sensor Failures - see "Chart 6" on page 34250241.
(3) TAWS - see "Chart 8" on page 34250247.
(4) GDU 1050 - see "Chart 9" on page 34250249, "Chart 13" on page 34250269, and "Chart 14" on page 34250271.
(5) GDU 1250A - see "Chart 9" on page 34250249, "Chart 13" on page 34250269, and "Chart 14" on page 34250271.
(6) GCU 47X - see "Chart 15" on page 34250281.
(7) GMA 350c - see "Chart 16" on page 34250283 and "Chart 17" on page 34250284.
(8) GIA 64(W) - see "Chart 18" on page 34250288 and "Chart 19" on page 34250290.
(9) GEA 71 - see "Chart 20" on page 34250299.
(10) GTX 3X5R - see "Chart 21" on page 342502103.
(11) GDC 72 - see "Chart 22" on page 342502105.
(12) GTP 59 - see "Chart 5" on page 34250235.
(13) GRS 79 - see "Chart 23" on page 342502110 and "Chart 24" on page 342502111
(14) GMU 44(B) - see "Chart 23" on page 342502110 and "Chart 24" on page 342502111
(15) GDL 69eA / 69A SXM - see "Chart 25" on page 342502115 and "Chart 26" on page 342502116.
(16) FS 510 - see "Chart 27" on page 342502118.
(17) GSR 56 - see "Chart 28" on page 342502120.
(18) GWX 68 / 75 - see "Chart 29" on page 342502122.
(19) GTS 825 - see "Chart 30" on page 342502128.
(20) GMC 710 - see Chart 3 in 22-10-00.
(21) Software Configuration - see "Chart 33" on page 342502153.
(22) Synthetic Vision System - see "Chart 35" on page 342502169.

System Status Page (AUX Group)
Figure 14

(23) GFC 700 AFCS

See "Chart 7" on page 34250244.

The GFC 700 is a digital Automatic Flight Control System (AFCS) which is integrated into various components of the G1000 NXi. This section touches upon key items to note while troubleshooting the GFC 700.

Should a problem be encountered during the operation of the GFC 700, the pilot and technician should first evaluate the overall status and condition of the G1000 NXi system at the AUX – System Status page (on MFD). Any alert messages, annunciations, or other abnormal behaviors should be noted in an effort to pinpoint the fault. The object is to locate the fault within a LRU or LRUs in efforts to replace the defective equipment.

The GFC 700 AFCS Annunciation field is located above the airspeed tape on the PFD as shown in "Chart 7" on page 34250244.

(24) GFC Status Page

The GFC Status page is presented in configuration mode and gives status information regarding the GFC 700.

See "Figure 15".

(a) GIA Status

AP Disconnect: Shows the condition of the AP DISC +28 VDC input to the GIAs and servos, which is required for the Autopilot to operate. A green status indicator shows the AP DISC switch is closed and the GFC 700 is actively receiving 28. volts. A black indicator box indicates the GIAs and servos are no longer receiving the +28 VDC AP DISC power (switch open or other fault).

(b) Monitor/Control Board Status

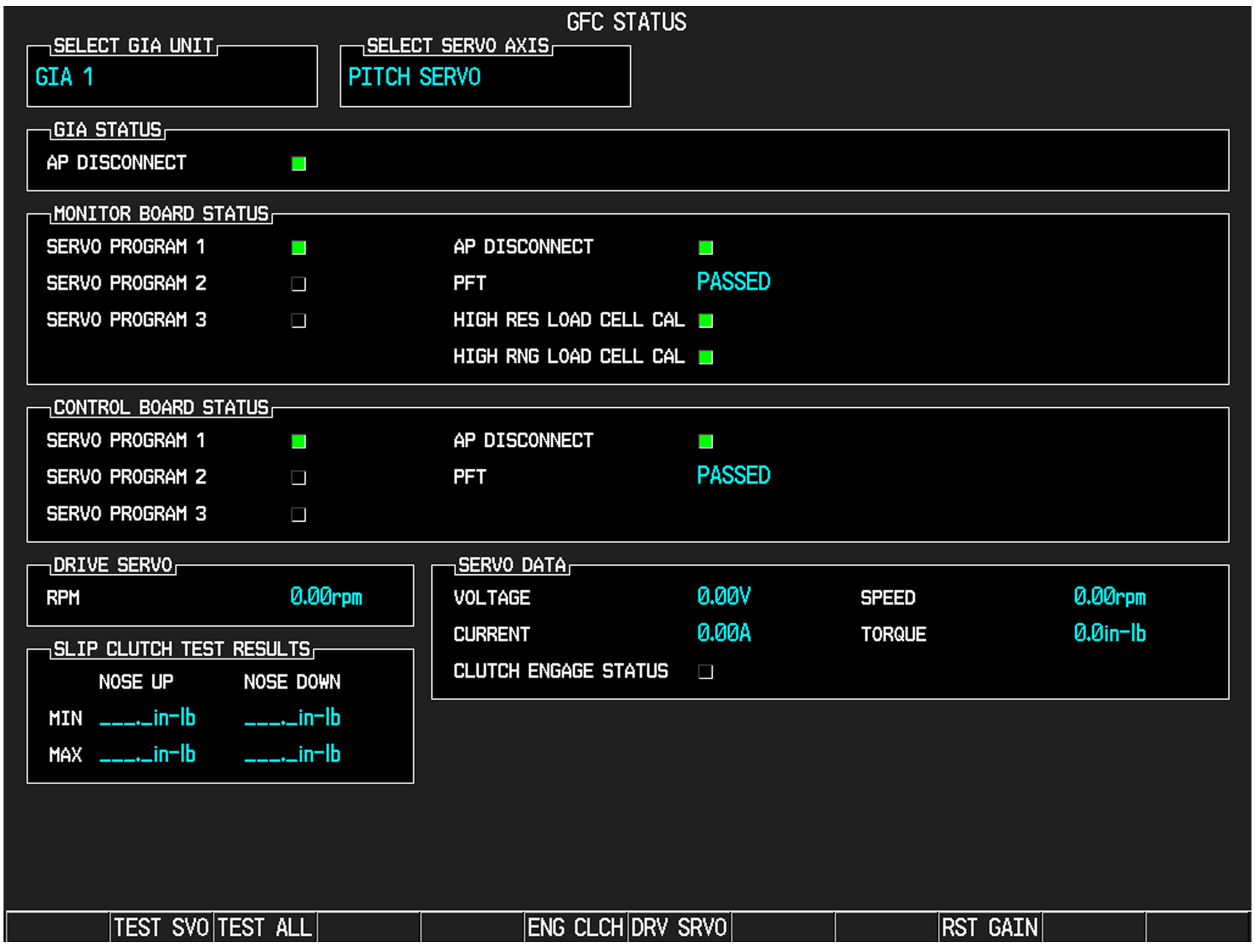
Shows the condition of various monitor board components.

(c) Servo Program (1-3): Servo program discretes are used to determine the HW strapping for each GSA to define the servo type. This information can be cross-referenced against the system interconnects to verify proper servo grounding.

(d) AP Disconnect: Same as GIA Status.

(e) PFT: Indicates whether the pre-flight test has passed or failed.

(f) HIGH RES & HIGH RNG LOAD CELL CAL: Shows the condition of the high resolution and high range load cells on the monitor board. If box is black, this indicates a corrupt or missing load cell calibration; return the servo to Garmin.



GFC Status Page
Figure 15

(25) Drive Servo

Allows the technician to enter a desired RPM at which to manually drive the selected servo. Direction of rotation is controlled by the polarity of the RPM (+ or -). After the speed is entered, the technician may use the ENG CLCH and DRV SRVO softkeys to drive the servo.

NOTE: Be especially certain that the flight controls are clear and safe to operate before manually driving the servo.

SERVO DATA

Shows real-time reported data including servo voltage, speed, motor current, load cell torque, and clutch solenoid status. A green box indicates the servo clutch is engaged.

(26) GIA Fault Log Descriptions

The section was created to help determine why the GFC 700 has failed the Pre-Flight Test (PFT) indicated by the red PFT annunciation, it defines the PFT sequence for the servos and the GFC 700 system and then provides troubleshooting information to help resolve failures.

There are 16 steps to the GFC 700 PFT. The PFT is performed by both GIA's at startup, and needs to pass on both GIA's before the autopilot can be engaged.

The PFT is only started if the AHRS has aligned, the GIA's and servos are configured and the certification gains are valid. If the PFT has not completed after one minute from when the initialization started, the PFT will fail. After the system PFT has passed, it will be performed again if a servo resets, if the autopilot servo breaker is reset or the cross side GIA restarts it.

Generally, the PFT failure fault is logged in the GIA Maintenance Log and not in the Servo Maintenance Logs unless the GIA log fault identifies a servo problem.

NOTE: Thoroughly understanding the operation of the G1000 NXi system in Configuration mode is recommended before starting this procedure. The GFC Status page may be used to check the status of the servos and engage them to aid in troubleshooting.

(a) To access the GIA and GSA Maintenance Logs, perform the following steps:

1) Start the G1000 NXi in Configuration mode.
2) Use the FMS knob on PFD1 to go to the Diagnostics Terminal page in the System group. This page allows the technician to view maintenance logs associated with the GFC 700.
3) Choose 'GIA1' or 'GIA2' in the LRU window.
4) In the SERVO window, choose 'NONE' to view the GIA Maintenance Log, or choose a servo to view their logs.
5) Using the FMS knob, choose 'VIEW MAINTENANCE LOG' in the COMMAND window.
6) Press the ENT key.
7) When the Maintenance Log data starts to display in the OUTPUT window, you may see "More...press any key to continue..." at the bottom of the OUTPUT window. This informs you there is more data to display and the system has paused allowing you to view the data before continuing. To see more of the data, reselect the "VIEW MAINTENANCE LOG" in the COMMAND window and press the ENT key. The "...press any key to continue..." function is not active at this time.
8) Scroll through the OUTPUT list by pressing the OUTPUT softkey.

(b) The GIA Maintenance Log can record any of the following faults:

1) FCS Task not started: Bad gains.

The FCS task has not started because the gains are not present or have been corrupted.

Reload the gain files to correct.

2) FCS Task not started: Gain structure out of range.

The FCS task has not started because the gains are not compatible with the GIA software. Reload the gain files to correct.

3) PFT FAIL: Timeout, <STEP>

Pre-flight test has failed because the specified step has not passed in the allotted time. See the GIA pre-flight steps for a description of the possible values for <STEP> on the failed GIA and corrective actions.

4) PFT FAIL: Cross GIA Failed, State: <STEP>

Pre-flight test has failed on opposite GIA. <STEP> specifies the pre-flight test step on selected GIA that was in progress when the pre-flight test failed on the opposite GIA. See the GIA pre-flight steps for a description of the possible values for <STEP> on the failed GIA and corrective actions.

5) PFT FAIL: <STEP>

Pre-flight has failed because the step specified has failed. See the GIA pre-flight test steps for a description of the possible values for <STEP> on the failed GIA and corrective actions.

6) AHRS MON invalid: <STATE>

The AHRS monitor has detected that the AHRS data is invalid. The possible values for <STATE> are:

a) Mon Prmtr Invalid: The ARINC 429 data used by one of the monitors has not been received.

b) Attitude Prmtr Invalid: The ARINC 429 pitch or roll angle has not been received.

c) Exceeded Attitude Limits: The pitch or roll angle has exceeded the engagement limits.

d) Cross Hdg Accel Fail: Cross heading acceleration monitor failed.

e) Vert Accel Fail: Vertical acceleration monitor failed.

f) Fltrd Cross Hdg Accel Fail: Filtered cross heading acceleration monitor failed.

g) Fltrd Vert Accel Fail: Filtered vertical acceleration monitor failed.

h) Roll Accel Fail: Roll acceleration monitor has failed.

i) Normal Accel Fail: Normal acceleration has failed.

7) Troubleshoot the GRS 79 for the cause of the failure.

8) Stuck switch invalidated parameter: <AXIS>

A MET switch in the specified axis is stuck. Check the MET (trim) switches for proper operation.

9) PRMTR: <PARAMETER> MODE:<MODE> Parameter lost

The mode specified by <MODE> has been disengaged because the parameter specified by <PARAMETER> has become invalid. The following is a list of some of the possible values for <PARAMETER>:

10) AD TDM Comm Valid: The specified mode has been disengaged because communication with the servos, via the Time Division Multiplexer protocol, has been lost.

11) AP Pitch MET not stuck: The specified mode has been disengaged due to a stuck pitch MET switch.

12) Check the MET (trim) switches for proper identification.

(c) GIA Pre-Flight Test Steps

0) PFT step 0: System initialization, verify GFC powered.

This step is checking to make sure the GFC is powered up. It checks to make sure the GIA AP disconnect input is connected to 28 volts, and makes sure the Servos are up and communicating. If this step fails, make sure the GIA is connected to AP disconnect by using the GFC configuration page. Also make sure all configured Servos are communicating by checking for Servo product data in configuration mode.

1) PFT step 1: System initialization, verify GIA audio is valid.

This step is checking to make sure that the GIA audio region has been loaded and configured. If this step fails, load GIA audio region and audio configuration.

2) PFT step 2: System initialization, verify required servos are configured.

This step is checking to make sure the current Servo configuration matches the Servo configuration specified in the certification gain file. If this step fails, then make sure the Servo configuration on the GFC configuration page matches the Servo configuration specified in the configuration gain (.cgn) file.

3) PFT step 3: System initialization, verify selected side.

This step is checking to make sure the PFD1 is online and sending the selected AFCS side data over.

HSDB to GIA1. If this step fails, then make sure the PFD is powered up and there is an Ethernet connection from the PFD to the GIA.

4) PFT step 4: System initialization, verify AHRS monitor

This step is checking to make sure the AHRS monitor is valid and not reporting an AHRS failure.

NOTE: AHRS monitor will be assumed valid if on the ground. If this step fails, then make sure the AHRS and ADC is powered up and sending valid attitude data to the G1000 NXi.

5) PFT step 5: System initialization, verify servo PFT is complete.

This step is checking to make sure that all servos have completed their own PFT. This does not check whether the servo PFT passed or failed. It verifies that the servo PFT is no longer in progress.

6) PFT step 6: Verify cross GIA is initialized.

This step is checking to make sure the other GIA is also on step 6 of its PFT. If this step fails, try cycling power on GIA1, GIA2, and all servos. If PFT still fails at step 6 then you will need to go check the PFT status of the other GIA and see what step it is failing (it should be prior to step 6).

NOTE: The PFT status is communicated between GIA1 and GIA2 using HSDB. As a result, both the PFD and MFD must be powered for this state to pass.

7) PFT step 7: Verify servo type.

This step is checking to make sure the Servos are the correct type (80 or 81, high or low torque). If this step fails, make sure the Servo configuration on the GFC configuration page matches the Servos installed in the aircraft.

8) PFT step 8: Verify servo first certification data.

This step is checking to make sure the servos have the same certification gains loaded in them as the GIAs have. If this step fails, reload the certification gains in GIA1, GIA2, and all servos.

9) PFT step 9: Verify servo second certification data.

This step is checking to make sure the servos have the same certification gains loaded in them as the GIAs have. If this step fails, reload the certification gains in GIA1, GIA2, and all servos.

10) PFT step 10: Updating servo RTC.

This step is setting the system time in the servos to by the same time as the GIA system time. This step should never fail.

11) PFT step 11: Verify servo PFT status.

This step is checking to make sure the servos have all passed their own PFT. If this step fails, please proceed to servo PFT explanation below.

12) PFT step 12: Verify AP disconnect enabled.

This step is checking to make sure GIA1, GIA2, and all servos have are connected to a 28 volt AP disconnect. If this step fails, make sure the AP disconnect input to GIA1, GIA2, and all servos is connected and registering 28 volts. Make sure the AP disconnect switch has been installed in the aircraft. Make sure no one is holding the AP disconnect switch down on the yoke.

13) PFT step 13: Verify servo validity.

This step is checking to make sure all the Servos are up and communicating with valid data to the GIA in TDM mode. If this step fails, then make sure all Servos are turned on and communicating by checking for green boxes on the GFC configuration page.

14) PFT step 14: Verify cross GIA PFT is completed.

This step is checking to make sure the other GIA is also on step 14 of its PFT. If this step fails, try cycling power on GIA1, GIA2, and all servos. If PFT still fails at step 14 then you will need to go check the PFT status of the other GIA and see what step it is failing.

NOTE: The PFT status is communicated between GIA1 and GIA2 using HSDB. As a result, both the PFD and MFD must be powered for this state to pass.

15) PFT step 15: PFT completed.

The PFT has successfully completed.

16) PFT step 16: PFT failed.

The PFT has failed.

(d) Servo Faults and Troubleshooting

Whenever a servo fault occurs, a status message is logged to the corresponding servo control or monitor maintenance log. This information is also accompanied by a time and date stamp. An “RTC DATE” entry is made every time a servo is powered on, it is normally not useful for troubleshooting.

The following is a listing of possible faults that could be reported in a GSA fault log. Faults can occur in either the monitor board processor or the control board processor, both of which are contained in the GSA unit.

1) Monitor Processor

The monitor processor contains the logs that are found in these processors -

2 – Pitch Servo

4 – Roll Servo

6 – Yaw Servo

8 – Pitch Trim Servo

There are two main groupings of faults that can occur in the monitor processor:

a) The first grouping of faults can occur during the GSA unit pre-flight test (PFT). If there is a fault during PFT the unit will not be able to transition to normal mode and the only way to clear this state would be to cycle unit power.

b) The second grouping of faults can occur during normal mode. These faults generally cause a disconnect of power to the GSA and report that a fault has occurred to the GIA. The Notes column indicates any actions that can be taken to troubleshoot the problem in the aircraft by the technician. Any faults that are not listed here indicate an internal problem requiring replacement of the servo. If the items in the Notes column check out ok, replace the servo.

2) Control Processor

The control processor contains the logs that are found in these processors –

3 – Pitch Servo

5 – Roll Servo

7 – Yaw Servo

9 – Pitch Trim Servo

There are two main groupings of faults that can occur in the control processor.

a) The first grouping of faults can occur during the GSA unit pre-flight test (PFT). If there is a fault during PFT the unit will not be able to transition to normal mode and the only way to clear this state would be to cycle unit power.

b) The second grouping of faults can occur during normal mode. These faults generally cause a disconnect of power to the GSA and report that a fault has occurred to the GIA.

The Notes column indicates any actions that can be taken to troubleshoot the problem in the aircraft by the technician. Any faults that are not listed here indicate an internal problem requiring replacement of the servo. If the items in the Notes column check out OK, replace the servo.

CHART 1
MONITOR PROCESSOR- PFT FAULTS

MONITOR PFT STEP	NOTES
"INTERNAL COMM FAIL"	This can sometimes be a result of a failure on the other internal servo board, check faults on the other processor
"UNSW POWER INV"	Check unit power
"MON SOL PWR ON FAIL"	Check unit power and AP Disconnect power
"CTL SOL PWR ON FAIL"	Check unit power and AP Disconnect power
"SOL PWR FAIL"	Check unit power and AP Disconnect power
"CERT DATA UNINSTALLED"	Upload the certification gain file to the Monitor board
"STRAP CODE MISMATCH"	Check the connector strap inputs to the unit

CHART 2
MONITOR PROCESSOR - NORMAL MODE FAULTS

MONITOR FAULT	NOTES
"GIA DIS FAULT"	Check the AP Disconnect power into the unit
"HOST DATA DIF"	Check the AHRS wiring to the system
"HOST DATA INV"	Check the AHRS wiring to the system
"SVO PWR INV"	Check unit power and AP Disconnect power
“STRP CODE CHNG”	Check the connector strap inputs to the unit
“MET STUCK SWTCH”	Check the MET switch inputs into the system
“MET STATUS DIF”	Check the MET switch inputs into the system

(27) Downloading GIA and GSA Maintenance Logs

If additional assistance is needed troubleshooting autopilot faults, the Maintenance logs can be downloaded to an SD card as a text file (.txt) and emailed to Garmin Aviation Product Support. Please call Garmin Aviation Product Support before you send a Maintenance Log to notify them you are sending it to prevent a delay in response. You may download multiple GIA and GSA Maintenance Logs to the same file, however in your email to Garmin you must furnish the order in which they were downloaded (i.e. GIA1, then GIA2, then SRVO PTCH MON, then SRVO PTCH CTL, etc.).

(a) Insert a FAT 32 formatted SD card into the top slot of the PFD before turning on the displays.

(b) Power up PFD1/2 and MFD in the configuration mode.

(c) On the PFD1 in the System page group, use the small FMS knob to scroll to the Diagnostics Terminal page.

(d) Press the LG2CRD softkey at the bottom of the PFD1. Verify that the softkey text grays out. This indicates the recording function is active and all text that is displayed in the OUTPUT window will be saved to the card.

CHART 3
CONTROL PROCESSOR - PFT FAULTS

CONTROL PFT STEP	NOTES
"INT COMM TEST FAIL"	This can sometimes be a result of a failure on the other board, check faults on other processor
"CTL MOT PWR ON FAIL"	Check unit power and AP Disconnect power
"MON MOT PWR ON FAIL"	Check unit power and AP Disconnect power
"HALL 1 FAIL"	Check unit power and AP Disconnect power
"HALL 2 FAIL"	Check unit power and AP Disconnect power
"HALL 3 FAIL"	Check unit power and AP Disconnect power
"HALL 4 FAIL"	Check unit power and AP Disconnect power
"HALL 5 FAIL"	Check unit power and AP Disconnect power
"HALL 6 FAIL"	Check unit power and AP Disconnect power
"CURR OFFST FAIL"	Check unit power and AP Disconnect power
"SVO TYPE FAIL"	Check unit power and AP Disconnect power
"CERT DATA UNINSTALLED"	Upload the certification gain file to the Control board
"STRAP CODE MISMATCH"	Check the connector strap inputs to the unit

CHART 4
CONTROL PROCESSOR - NORMAL MODE FAULTS

CONTROL FAULT	NOTES
"GIA DIS FAULT"	Check the AP Disconnect power into the unit
"HOST DATA DIF"	Check the AHRS wiring to the system
"HOST DATA INV"	Check the AHRS wiring to the system
"SVO PWR INV"	Check unit power and AP Disconnect power
“STRP CODE CHNG”	Check the connector strap inputs to the unit
“MET STUCK SWTCH”	Check the MET switch inputs into the system
“MET STATUS DIF”	Check the MET switch inputs into the system

(e) Enable the curser by pressing the FMS knob, select “GIA1” in the LRU drop down menu and then press the ENT key to select it.

(f) Skip the SERVO box and move the cursor to the COMMAND box and select “View Maintenance Log” in the drop down menu then press the ENT key. The error log data will be displayed in the OUTPUT box. If you see the “more…press any key to continue” text at the bottom of the screen, you may need to reselect “View Maintenance Log” for GIA data to allow it to continue scrolling down the screen (pressing any key will not continue, disregard the text instruction you to do so). Continue to scroll through all the OUTPUT data until you see the text, “End of Fault Log”.

(g) Move the curser back to the LRU box, select “GIA2” in the LRU drop down menu and then press the ENT key to select it.

(h) Skip the SERVO box and move the cursor to the COMMAND box and select “View Maintenance Log” in the drop down menu then press the ENT key. The error log data will be displayed in the OUTPUT box. If you see the “more…press any key to continue” text at the bottom of the screen, you may need to reselect “View Maintenance Log” for GIA data to allow it to continue scrolling down the screen (pressing any key will not continue, disregard the text instruction you to do so). Continue to scroll through all the OUTPUT data until you see the text, “End of Fault Log”.

(i) If you need to download Servo fault logs (usually done at the request of Garmin Product Support), perform the following steps. Otherwise, skip to step (j).

1) In the LRU box, you may select either “GIA1” or “GIA2”.
2) In the SERVO box, choose a servo using the FMS knobs. Each servo contains two logs, one in the Monitor (MON) processor and one in the Control (CTL) processor. You must download both for each servo separately.
3) In the COMMAND box, select “View Maintenance Log” and press the ENT key.
4) The log will appear in the OUTPUT box. It will scroll to the end automatically. When it is complete, repeat steps 1) - 3) for the other servos in the aircraft. Be sure to note the order the servos were downloaded in including the Monitor or Control logs to email to Garmin Product Support. Without knowing the order in which the logs were downloaded, Garmin will be unable to process them and will ask for another full download.

(j) Press the LG2CRD softkey to turn off the recording function.

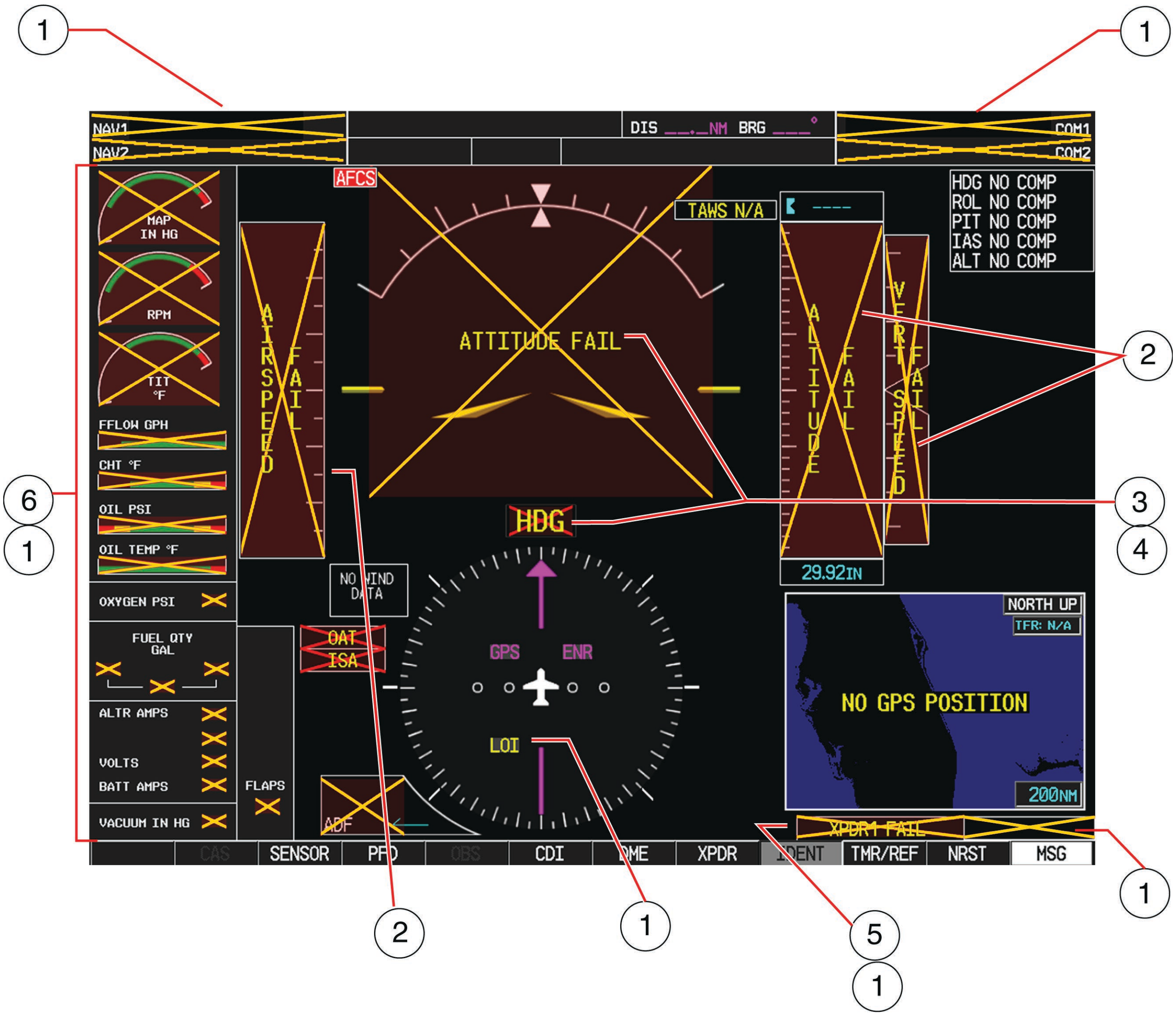
(k) Wait 1 minute for the system to save the data from the download to the SD card.

(l) While you are waiting for the data to be saved to the SD card, record the order of the LRU’s and/or Servos were downloaded so that you can provide that information to Garmin to help decipher the order of the error data.

(m) Power down the G1000 NXi System and remove the SD card.

(n) Insert the SD card in the card reader of a laptop or desktop computer and open the “diag_buf_log.txt” file from the SD card using the WordPad program. Verify that all of the fault logs were downloaded by checking for the “End of Fault Log” message at the end of the GIA data, and that the last servo log entry has the current date.

(o) Insert the fault log as an attachment to an email and include the LRU order how the data was downloaded and send to Garmin Aviation Product Support at avionics@Garmin.com.



1. GIA 64(W) Integrated Avionics Units
2. GDC 72 Air Data Computer
3. GRS 79 AHRS
4. GMU 44(B) Magnetometer
5. GTX 3X5R Transponder
6. GEA 71B Engine Airframe Units

System Fail
Figure 16

CHART 5 (Sheet 1 of 6)
LRU TROUBLESHOOTING

System Failure Troubleshooting

The following table provides basic troubleshooting guidance for LRU failures.

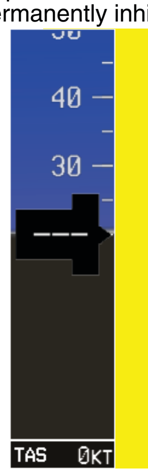
Annunciation	Associated LRU(s)	Solution
Low Speed Awareness Band permanently displayed 	GDU	• With aircraft weight on wheels and the G1000 NXi in Configuration Mode, check that the **GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is illuminated. • If indication is illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace defective unit. ✓ If indication is not illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
Low Speed Awareness Band permanently inhibited 	GDU	• With aircraft weight off wheels and the G1000 NXi in Configuration Mode, check that the **GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is not illuminated. • If indication is not illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace unit. ✓ If indication is illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
NAV1 & COM1 = GIA1 	GIA1	• Check configuration settings for GIA1 and PFD1. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA1, replace GIA1. • Check Ethernet interconnect from GIA1 to PFD1. ✓ If problem persists, replace PFD.
NAV2 & COM2 = GIA2 	GIA2	• Check configuration settings for GIA2 and PFD2. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA2, replace GIA2. • Check Ethernet interconnect from GIA2 to PFD2. ✓ If problem persists, replace PFD.

CHART 5 (Sheet 2 of 6)
LRU TROUBLESHOOTING

Annunciation	Associated LRU	Solution
GPS DR  	GIA1 or 2	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check PFD1 Alert Window for GIA1/2 configuration, software or failed data path error messages. Correct any errors before proceeding. • Verify the aircraft is located where the GPS antennas have a clear view of the sky. • Verify the aircraft is not parked in close proximity to a hanger with the doors open equipped with a GPS repeater. • On the MFD AUX-GPS STATUS page, check for erratic GPS Signal Strength bars. If they are erratic, external interference is affecting the GPS receiver. Locate source of interference and remove. • Swap GIA1 and GIA2 to verify location of problem. If problem follows the GIA, replace the GIA. • Check corresponding GPS antenna and cable for faults. Correct antenna or cable fault. • Check PFD1 to GIA1 and PFD2 (or MFD) to GIA2 Ethernet interconnect for faults. Correct interconnect fault. • If problem persists; replace PFD1, PFD2 or the MFD that shows the problem.

CHART 5 (Sheet 3 of 6)
LRU TROUBLESHOOTING

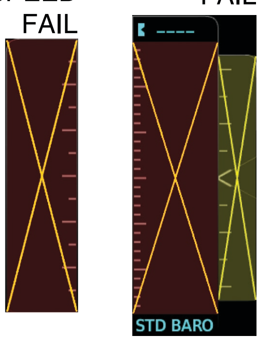
Annunciation	Associated LRU	Solution
XPDR FAIL 	GTX 3X5R	• Check PFD Alert Window for GIA1/2 and GTX 3X5R configuration, software or failed data path error messages. Correct any errors before proceeding. • Perform a SET>ACTV configuration reset on the GTX Transponder Configuration page for each installed GTX. • For GTX 3X5R transponders verify the aircraft registration is entered in the GTX Transponder Configuration page. • Check the GIA and GTX racks for connector pin faults (push-back or bent) on the RS-232 interconnect lines. • Replace the GTX 3X5R.
AIRSPEED FAIL ALTITUDE VERT SPEED FAIL 	GDC 72	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check PFD1 Alert Window for PFD1/2, MFD or GDC configuration, software or failed data path error messages. Correct any errors before proceeding. • Inspect GDC 72 pitot/static ports and plumbing for blockage. • Check GDC 72 configuration settings for the PFDs, MFD, GIA1, and GIA2. Reload if unsure they are correct. • If PFDs, MFD, and GIA configuration settings are correct, replace the GDC 72. • If problem persists, replace the GDC 72 configuration module.
OAT 	GTP 59	• Check OAT probe wiring, probe and connectors for faults or damage. • Replace GDC 72 config module and pigtail harness. Replace the GTP 59. • If problem remains, replace GDC 72.

CHART 5 (Sheet 4 of 6)
LRU TROUBLESHOOTING

Annunciation	Associated LRU	Solution
HDG FAIL 	GRS 79 & GMU 44(B)	• Check PFD1 Alert Window for PFD1/2, MFD or GRS configuration, software or failed data path error messages. Correct any errors before proceeding. • Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements. • Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power after moving aircraft away from metal objects to determine if metal objects were the source of the interference. Allow up to five minutes for the heading to reinitialize. • Perform a Magnetometer Interference Test to check for interference from onboard electrical system components (e.g. NAV lights). Pay particular attention to any new electrical devices that have been installed since the aircraft was new. Correct any discrepancies that do not allow this test to pass before continuing. • Ensure GRS 79 and GMU 44(B) connectors are secure. • Check the wiring and any inline connectors between the GRS and GMU for faults. • Recalibrate the GMU 44(B). • Replace the GMU 44(B). • If problem persists, replace the GRS 79.
	GRS 79 & GMU 44(B)	• If this message persists longer than five minutes, perform AHRS calibration procedures as described later in this chapter.

CHART 5 (Sheet 5 of 6)
LRU TROUBLESHOOTING

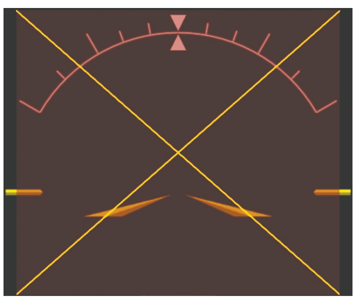
Annunciation	Associated LRU	Solution
ATTITUDE FAIL 	GRS 79	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements. • Check PFD1 Alert Window for PFD1/2, MFD or GRS configuration, software or failed data path error messages. Correct any errors before proceeding. • Ensure GRS 79 unit connecter is secure and proper wire harness strain relief is provided. • Ensure the GRS 79 is fastened down tightly in its mounting rack and that the mounting rack is not loose (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Cycle GRS 79 power to restart initialization. • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important for an ATTITUDE FAIL that appears during ground operation only. • Perform an Engine Run-Up Test to check if engine vibration is causing the GRS 79 to go offline. • Replace GRS 79. • If problem persists, replace the GRS79 configuration module. • Contact Garmin Aviation Product Support if condition continues after replacing the GRS 79 and config module for additional assistance.

CHART 5 (Sheet 6 of 6)
LRU TROUBLESHOOTING

A DME or ADF failure is represented by the following red X's. The PFD displays function as a control head for the remote-mounted ADF and DME.

Annunciation	Sensor	Solutions
ADF FAIL OR NO ADF SIGNAL  GIA2 FAIL 	ADF	• Ensure that GIA #2 is properly functioning. • Reload the ADF option configurations. • Check for proper operation of the ADF receiver. Ensure that the receiver is receiving power. • Check ADF – GIA2 interconnect.
 	DME	• Ensure that GIA #2 is properly functioning. • Reload the DME option configurations. • Check for proper operation of the DME receiver. Ensure that the receiver is receiving power. • Check DME – GIA2 interconnect.

CHART 6 (Sheet 1 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

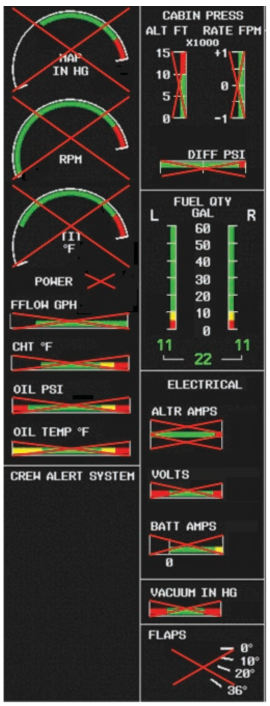
Invalid Data Field	Associated LRU(s)	Solution
Engine/Airframe Sensors (All Invalid)  (Typical)	GEA 71B & GIA 64(W)	• If software was loaded to a new GDU display, be sure that the user settings for the replaced display were cleared. Clear user settings by pressing the CLR key on the replaced display while applying power to it. Acknowledge the on-screen prompt by pressing the ENT key or the right-most softkey. • Check for GEA related Alert messages on the PFD. Correct any Alerts concerning software or configuration errors by reloading software or configuration as noted. • Verify GEA internal power supply, configuration, and calibration status in configuration mode. ✓ The internal power supply, configuration and calibration boxes should be green. If they are red, replace the GEA 71B. • Verify internal, external, and reference voltages listed in the Main Analog and I/O A Analog boxes are not dashed out (does not include Aircraft Power 1 and 2). ✓ If any voltages are dashed out, replace the GEA. • Ensure the GEA is online (green checkmark on the AUX – SYSTEM STATUS page). ✓ If GEA is not online verify unit is receiving power at the rack connector. ✓ Check the GIA/GEA interconnects for faults. ✓ Reload configuration files to both GIA's and the GEA. • If problem persists, replace the GEA 71B.

CHART 6 (Sheet 2 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Engine/Airframe Instrument Failures

The following table provides guidance for troubleshooting individual engine/airframe sensor failures. Be sure to also follow previous guidance given for the GEA 71B. The technician should troubleshoot to isolate the fault by checking sensor-to-GEA wiring, replacing the suspect sensor, and finally by replacing the GEA 71B. Replace one part at a time. Check for correct operation of the sensors and GEA 71B after any part has been replaced. Refer to G1000 NXi/GFC 700 in 91-00-00 Wiring Diagrams.

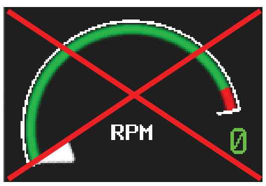
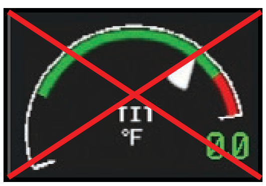
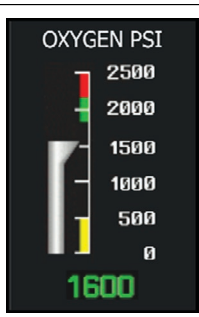
Invalid Field	**Sensor**	**Possible Solutions** (for applicable engine/system)
	Manifold Pressure	• Check manifold pressure transducer wiring. • Replace manifold pressure transducer wiring. • Replace GEA 71B.
	Propeller RPM	• Check engine tachometer sensor wiring. • Replace engine tachometer sensor. • Replace GEA 71B.
	Turbine Inlet Temperature	• Check TIT probe wiring. • Replace TIT probe sensor. • Replace GEA 71B.
	Fuel Flow	• Check fuel flow transmitter wiring. • Check fuel flow transmitter - GEA wiring. • Check power input to fuel flow transmitter. • Replace fuel flow transmitter. • Replace GEA 71B.
	Oil Pressure	• Check oil pressure sensor – GEA wiring. • Replace oil pressure sensor. • Replace GEA 71B.
	Oil Temperature	• Check oil temperature sensor – GEA wiring. • Replace oil temperature sensor. • Replace GEA 71B.
	Oxygen (Matrix Only)	• Check oxygen pressure transducer wiring. • Replace oxygen pressure transducer sensor. • Replace GEA 71B.

CHART 6 (Sheet 3 of 3)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

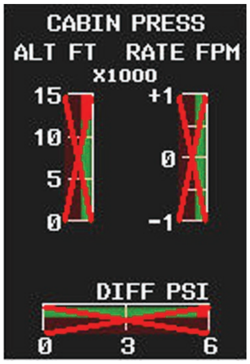
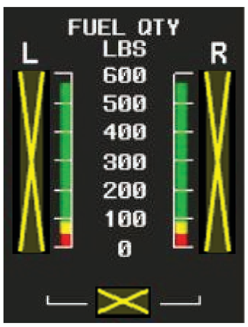
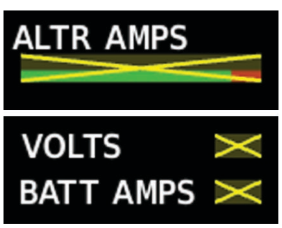
Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Cabin Pressure (Mirage Only)	• Check GAE 43 sensor – GEA 71B wiring. • Replace GAE 43 sensor. • Replace GEA 71B.
	Differential PSI (Mirage Only)	• Follow cabin pressure steps. • Follow GDC 72 steps. • Check GEA 71B - GIA 64(W) wiring. • Check GDC 72 - GIA 64(W) wiring. • Replace GEA 71B. • Replace GDC 72.
	Fuel Quantity	• Check Fuel Senders - #1 GIA 64(W) wiring. • Check #1 GIA 64(W) - #1PFD wiring. • Check MFD - #1 PFD connections. • Replace Fuel Senders left or right. • Replace #1 GIA 64(W). • Replace #1 PFD. • For more details see 28-00-00.
	Amp/Volts	• Check Alt amp sensor - GEA 71B wiring. • Replace Alt amp sensor. • Replace GEA 71B. • Check main bus - GEA 71B wiring. • Check emergency bus - GEA 71B wiring. • Replace GEA 71B.
	Vacuum (Mirage - Std) (Matrix - w/ Deice Opt)	• Check vacuum sensor - GEA 71B wiring. • Replace vacuum sensor. • Replace GEA 71B.
	Flaps	• Check Potentiometer - GEA 71B wiring. • Replace Potentiometer. • Replace GEA 71B.

CHART 7 (Sheet 1 of 3)
AFCS ANNUNCIATION TROUBLESHOOTING

GFC 700 ANNUNCIATION WINDOW

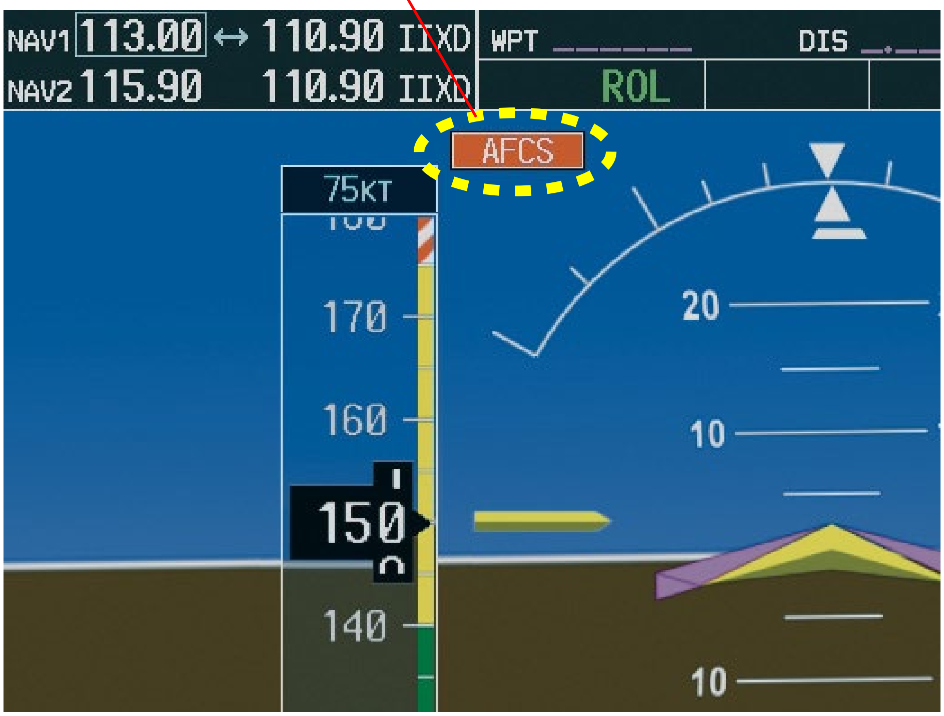


CHART 7 (Sheet 2 of 3)
AFCS ANNUNCIATION TROUBLESHOOTING

Condition	Annunciation	Recommended Actions
Pitch Failure	PTCH	• Check the AUX– SYSTEM STATUS page to see if the servo is online (green check). • Check that the affected servo is receiving power. • Check the servo wiring and connector. • Ensure PTRM switches are not stuck. • If failure condition still exists, remove and replace the affected servo.
Roll Failure	ROLL	
Yaw Damper Failure	YAW	
MEPT Sw. Stuck, or Pitch Trim Axis Control Failure	PTRM	
AFCS System Failure	AFCS	• Check that no red X’s are present on the MFD and PFDs. • Check that there are no Alert Messages present in the PFD Alert window. Correct any software or configuration errors noted. • Go to the AUX SYSTEM STATUS page on the MFD and verify that all LRUs have a ‘green’ check. • Download GIA fault logs (see “GIA Fault Log Descriptions”) and review for failure information. • If the OAT and TAS is Red-X’d and the attitude indication is present, troubleshoot per Chart 3.
Emergency Descent Mode (PA-46-350P)	EDM	• Decouple autopilot.
Elevator Mistrim Up	↑ELE	• If mistrim annunciations persist, check the Pitch and Pitch Trim servos for proper operation. Verify that the servo is online at the AUX - SYSTEM STATUS page. • Check the Pitch Trim servo wiring and connector. Ensure the servo is receiving power. • Check the aircraft trim control rigging. If mistrim condition still exists, remove and replace the affected servo.
Elevator Mistrim Down	↓ELE	
Aileron Mistrim Left	←AIL	• Check for possible fuel imbalance. • Check aileron control rigging. If mistrim condition still exists remove and replace the roll servo.
Aileron Mistrim Right	AIL→	
Rudder Mistrim Left	←RUD	• Check the AUX– SYSTEM STATUS page to see if the servo is online (green check). • Check that the affected servo is receiving power. • Check the servo wiring and connector. • If failure condition still exists, remove and replace the affected servo.
Rudder Mistrim Right	RUD→	
Preflight Test	PFT PFT	Reset system power. Allow the system to complete pre-flight tests. The preflight test should finish within 2 minutes. If it does not pass, the red ‘PFT’ annunciation is shown. In case of PFT failure, troubleshoot in the same manner as for the red ‘AFCS’ annunciation.

CHART 7 (Sheet 3 of 3)
AFCS GENERAL TROUBLESHOOTING

Condition	**Resolution**
Poor AP Performance (Limited AP Authority)	• Check each servo mount slip clutch per 22-10-00. Verify the clutches are not excessively loose and are within torque limits. • Check aircraft controls for proper balancing and adjustment per 55-20-00, 55-40-00, and 27-00-00. • Check aircraft control cables for proper tension per 22-10-00.
AP DISC Problems	• For intermittent nuisance disconnects with no AFCS or PFT alert, check A/P disconnect switch and wiring for intermittent faults. • If an AFCS or PFT alert is displayed at the time of the disconnect, troubleshoot per 22-10-00, Figure 9 and Post-Installation Procedures. • Check the GIA AFCS Fault Logs. If a **"Mon Prmtr Invalid"** message is received, check for valid true airspeed. A faulty GTP 59 OAT probe may cause TAS to become invalid, which will flag the "**Mon Prmtr Invalid"** message. • Contact Garmin Product Support for assistance.
Loss of Manual Electric Trim	• Check pitch trim servo status. • Check MET switch discrete inputs to both GIAs by going to the GIA I/O Configuration page and selecting DISCRETE IN inputs.
AutoTrim Inoperative	• If DATA indicator fails to illuminate or illuminates incorrectly, troubleshoot flap motor & discrete input wiring.

CHART 8
TAWS TROUBLESHOOTING

Annunciation	Cause	Solution
TAWS FAIL	A TAWS system failure has occurred.	• If message occurred on the first power up after unlocking TAWS, cycle power to initialize TAWS. • Ensure each GDU contains the correct database data card. • Ensure the G1000 NXi is configured for TAWS: ✓ If the system is not configured for TAWS, reconfigure per 34-20-00, TAWS Unlock Procedure. • Verify GIAs are online. • Ensure a database or GDU SW mismatch has not occurred. ✓ If a mismatch has occurred, load correct database/software files or replace the terrain card.
TAWS TEST	TAWS system is currently being tested.	N/A – Test will be conducted up to two minutes.
TAWS INHB	TAWS system alerting is disabled	Enable TAWS system alerting by pressing the INHIBIT softkey on the MAP – TAWS page.
TAWS N/A	GPS system integrity not high enough to enable TAWS	• Ensure valid GPS position is received from the AUX – GPS STATUS page. • Check GPS antenna & associated coaxial cabling. • Troubleshoot GIAs.

(28) Backup Communications Path Checks

The G1000 NXi system architecture is designed with redundant communication ports for several LRUs so that critical information can continue to be displayed in the event of an equipment or wiring failure. Of most importance is flight attitude, heading, and air data information. The GRS 79 and GDC 72 each have four separate ARINC 429 data lines which are all capable of sending data to the displays. The GEA 71B, GTX 33/33D/3X5R, and GMA 350c each have two redundant serial communication paths for the same purpose. See "Figure 2" on page 3425025 for a basic G1000 NXi block diagram depicting this architecture. When troubleshooting, refer to the G1000 NXi/GFC 700 Wiring Diagrams listed in Chapter 91-00-00.

(29) Failed Path Messages

The following message indicates there is a data path connected to the GDU (PFD1, PFD2, or MFD) or the GIA (1 or 2) that has failed.

FAILED PATH – A data path has failed.

The FAILED PATH message is triggered by a timeout of any one digital channel. The channels that are checked are listed on these pages in config mode:

(a) GDU RS-232 / ARINC 429 CONFIG (PFD1/2, and MFD).

(b) GIA RS-232 / ARINC 429 CONFIG (GIA1 and GIA2).

(c) GIA CAN / RS-485 CONFIGURATION (GIA1 and GIA2).

NOTE: Once the FAILED PATH message has been triggered, it will remain on the list of messages until the next power cycle. This latching was implemented so that for intermittent failures, the message would remain at the end of the flight (to alert maintenance crew). Also, this keeps the crew from having to acknowledge message repeatedly in the case of intermittent failures.

The box next to each channel indicates the current status of the channel per the below:

Red = data path is known to be failed.

Black = data path status is unknown.

Green = data path is known to be good.

The applicable data paths can be verified by viewing the configuration mode pages listed below in "Chart 9" – "Chart 11" on page 34250251. See "Chart 31" on page 342502145 to navigate to a particular configuration page.

CHART 9
G1000 - TROUBLESHOOTING - DATA PATH FAILURES

Failure Message	Cause	Solutions
FAILED PATH – A data path has failed.	A communications data path status is not being received by the G1000 NXi.	Determine which data path has failed: See G1000 NXi backup path test below. Check wiring continuity for the failed path. Swap or replace the affected LRU.

CHART 10
FAILED PATH MESSAGES, GDU RS-232 / ARINC 429 CONFIG PAGE (PFD1/2, AND MFD)

SELECT UNITS = MFD / PFD1 / PFD2 / GIA1 / GIA2 – ARINC-429 / RS-232 / RS-485 Serial Config Pages

Indicator	Status
	SELECT UNIT/LRU data path is functioning correctly.
	SELECT UNIT/LRU data path is not functioning correctly. • Check GTC avionics status page for SELECT UNIT or LRU configuration or software error messages. Correct any errors before proceeding. • If applicable, check the SELECT UNIT/LRU interconnect wiring and unit connector pins for faults. • Replace LRU. • Reload SELECT UNIT config file. • If problem remains, replace SELECT UNIT. • Remove SELECT UNIT and verify power and ground are present at the appropriate connector. ✈ If power or ground is not present, troubleshoot aircraft wiring for faults. ✈ If power and ground are present, check SELECT UNIT unit and LRU connector for damaged or pushback pins. • Swap PFD1 and PFD2 to confirm if the problem is in the original PFD1. • Replace original PFD1 if box displays green check after swapping displays. • Ensure LRU connecter is secure and proper wire harness strain relief is provided. • Swap LRU #1 and LRU #2 to confirm if the problem is in the original LRU #1. • Replace original LRU #1 if box displays green check after swapping units.
	SELECT UNIT/LRU data path functionally is unknown. • Reload SELECT UNIT configuration file.

CHART 11 (Sheet 1 of 2)
DATABUS ACTIVITY CHECKS

UNIT	BUS ACTIVITY	INTERFACE
PFD 1	RS 232 CH 1 IN/OUT GCU47X	1P10001-44/26/45 to P4751-2/6/1
	RS 232 CH 2 IN/OUT GMC710	1P10001-46/30/47 to P7101-2/6/1
	ARINC 429 CH 1 IN HIGH SPEED GRS79 #1	1P10001-18/19 to 1P791-8/9
	ARINC 429 CH 2 IN LOW SPEED GDC72 #1	1P10001-16/17 to 1P721-21/22
PFD 2	ARINC 429 CH 1 IN HIGH SPEED GRS79 #2	2P10001-18/19 to 2P791-8/9
	ARINC 429 CH 2 IN LOW SPEED GDC72 #2	2P10001-16/17 to 2P721-21/22
MFD	RS 232 CH 1 IN/OUT GCU47X	3P10001-44/26/45 to P4751-4/8/3
	RS 232 CH 2 IN/OUT GMC710	3P10001-46/30/47 to P7101-4/8/3
	ARINC 429 CH 1 IN HIGH SPEED GRS79 #2	3P10001-18/19 to 1P791-33/19
	ARINC 429 CH 2 IN LOW SPEED GDC72 #2	3P10001-16/17 to 1P721-8/9
GIA 1	RS 232 CH 1 IN/OUT GDC72 #1	1P603-41/42/43 to 1P721-69/68/70
	RS 232 CH 3 IN GAE43 ALT ENCODER	1P603-47/48 to P727-1/5
	RS 232 CH 5 IN/OUT GTX33 #1 w/ TIS	1P603-53/55/54 to 1P3301-23/22/51
	RS 232 CH 6 IN/OUT GRS79 #1	1P603-56/57/58 to 1P791-51/32/52
	RS 232 CH 7 IN/OUT GMA350C	1P603-59/62 to P3502-18/17
	ARINC 429 CH 1 IN GDC72 #2	1P603-29/31 to 2P721-49/50
	ARINC 429 CH 2 IN GRS79 #2	1P603-33/35 to 2P791-49/50
	ARINC 429 CH 5 IN LOW SPEED GDC72 #1	1P603-12/13 to 1P721-49/50
	ARINC 429 CH 6 IN HIGH SPEED GRS79 #1	1P603-14/15 to 1P791-49/50
	ARINC 429 CH 8 IN CPCS	1P603-18/19 to P250-12/13
	RS 485 CH 1 IN/OUT GEA 1	1P603-23/24 to P701-5/6
	RS 485 CH 4 IN/OUT GFC700	1P603-4/6 to P832-J/T & P834-J/T 1P603-5/7 to P831-J/T & P833-J/T

CHART 11 (SHEET 2 OF 2)
DATABUS ACTIVITY CHECKS

UNIT	BUS ACTIVITY	INTERFACE
GIA 2	RS 232 CH 1 IN/OUT GDC72 #2	2P603-41/42/43 to 2P721-69/68/70
	RS 232 CH 3 IN GAE43 ALT ENCODER	2P603-47/48 to P727-2/5
	RS 232 CH 4 IN/OUT WX-500	2P603-52/50/ 51 to P5003-8/20/ P2-5
	RS 232 CH 5 IN/OUT GTX345R/GTX33D/GTX345R	2P603-55/53/54 to [1P3251-30/8/51 (GTX335R)]/[1P3301-24/25/58 (GTX33D)]/ [2P3251-31/9/52 (GTX345R)]
	RS 232 CH 6 IN/OUT GRS79 #2	2P603-56/57/58 to 2P791-51/32/52
	RS 232 CH 7 PULSE OXIMETER	2P603-59/61/62 to P1-7/3/8
	RS 232 CH 8 GSR56	2P603-63/64/65 to P561-12/14/13 (optional)
	ARINC 429 CH 1 IN GDC72 #1	2P603-29/31 to 1P721-49/50
	ARINC 429 CH 2 IN GRS79 #1	2P603-33/35 to 1P791-49/5
	ARINC 429 CH 5 IN LOW SPEED GDC72 #2	2P603-12/13 to 2P721-27/7
	ARINC 429 CH 6 IN HIGH SPEED GRS79 #2	2P603-14/15 to 2P791-27/7
	RS 485 CH 1 IN/OUT GEA 1	2P603-23/24 to P701-7/8
	RS 485 CH 3 IN ADF	2P603-27/28 to P3502A-12/31 (optional) 2P603-36/37 to P3502A-13/14 (optional)
	RS 485 CH 4 IN/OUT GFC700	2P603-4/6 to P832-S/E & P834-S/E 2P603-5/7 to P831-S/E & P833-S/E

CHART 12
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E. Recommended Tools

The following tools (or equivalent) are recommended to perform various maintenance tasks on the G1000 NXi system.

(1) Calibrated voltmeter capable of measuring 0-32 Volts DC.
(2) #2 Phillips Screwdriver.
(3) 3/32nd inch Hex Tool.
(4) Calibrated digital Level with 0.25 degrees of accuracy capability.
(5) Calibrated VHF NAV/COM/ILS ramp tester.
(6) Calibrated transponder ramp tester including Mode S capability for Mode S transponder equipped aircraft.
(7) Air Data Test Set (ADTS) capable of simulating altitude up to the aircraft's service ceiling.
(8) Headset/Microphone.
(9) Outdoor line-of-site to GPS satellite signals or GPS indoor repeater.
(10) Ground Power Unit (Capable of supplying 28 Vdc).
(11) Calibrated Flight Control Cable Tension Meter or equivalent.
(12) Servo Mount Slip Clutch Adjustment Tool – Garmin P/N T10-00110-01 and a 2 Amp, 24 V, DC Power Supply.
(13) Calibrated digital thermometer suitable for measuring ambient temperature.
(14) Laptop with RS232 emulation software.
(15) M22885/108T8234 extraction tool.
(16) An 0.060" 6-Spline wrench.
(17) WX-PA portable analyzer kit (If Stormscope is installed).
(18) Calibrated torque wrench capable of measuring 0 – 70 in/lbs.
(19) Standard sockets & wrenches (3/8", 9/16", and 13/16").

F. Inspections

(1) Annual / 100 Hour Inspection

Each 12 months or 100 hours time-in-service, whichever comes first:

(a) See Inspections under Garmin GFC 700 in 22-10-00.

(b) Perform the "G1000 NXi Checks" on page 342502177.

(c) Perform the "Avionics Cooling Fan Fail Annunciation Test" on page 342502176.

(d) For airplanes equipped with GDL 69eA / 69A SXM, activate the Stall Warning horn and verify that the XM audio is muted.

(2) 500 Hour Inspection

Perform a functional test of the PFD1, MFD, PFD2, and Avionics cooling fans as follows: (see 21-20-00 for cooling fan locations)

(a) Turn OFF all electrical components and avionics systems.

(b) Verify the master switch is ON.

(c) Pull the three or four (as equipped) avionics FAN circuit breakers. I.E., PFD1 FAN, PFD2 FAN, AV/MFD FAN, AV/PFD2 FAN, MFD FAN, and/or AV FANS.

(d) One at a time, push IN each FAN circuit breakers. Verify airflow from each fan by motor sound and pull OUT that circuit breaker before moving to the next.

(e) After checking each fan, turn OFF master switch and push IN all three FAN circuit breakers.

(3) 1000 Hour Inspection

Each 1000 hours time-in-service visually inspect each unit listed below for corrosion, damage, or other defects. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

(a) GDU 1050 PFD and GDU 1250A MFD

Remove the PFDs and MFD. Inspect the mounting surfaces and connectors for corrosion, heavy oxidation, or other damage.

(b) PFD, MFD, and Avionics Cooling Fans

Inspect PFD1, PFD2, MFD, and the Avionics cooling fans for accumulation of dust or other damage. Remove excess dust as required. Perform the "500 Hour Inspection" on page 34250265.

(c) GMU 44(B) Magnetometer

Remove the GMU 44(B) and inspect the mounting hardware and GMU 44(B) for corrosion or other damage.

(d) For the following components, inspect the individual unit(s), rack(s), and connectors for corrosion or other defects.

1) GMA 350c Audio Panel
2) GIA 64(W) Integrated Avionics Units (2 ea.)
3) GEA 71B Engine/Airframe Unit
4) GTX 3X5R (Standard)
   GTX 33/33D (ES) (Diversionary Transponder), GTX 3X5R (Optional)
5) GDC 72 ADC
6) GRS 79 AHRS
7) GDL 69eA / 69A SXM - Datalink (if installed)
8) GSR 56 Iridium Satellite Transceiver (if installed)
9) GTS 825 TAS (if installed)

(e) Perform G1000 NXi Equipment Electrical Bonding Check in 51-80-00.

(f) Perform visual checks on the shield terminations for any degradation.

G. Components

(1) General

Maintenance of the G1000 NXi system components is “On Condition” only. The following provides basic information regarding unit removal, installation, and troubleshooting. Basic guidance on loading / configuring software is provided under "Complete Software Load" on page 342502157. When removing and/or replacing any G1000 NXi component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G1000 NXi LRU, it is required that the technician verify the LRU software part numbers.

To check an LRU software part number and/or version:

(a) Start the G1000 NXi system in configuration mode as described in "Configuration Mode Overview" on page 342502141

(b) The System Status page shows a list of LRUs in the LRU window. Activate the cursor and highlight the LRU window.

(c) Use the FMS knob to scroll through the list in the window and select the desired LRU.

(d) The software part number and version is displayed in the DATA window. Compare this to the latest configuration shown in the appropriate Required Equipment List (REL), "Chart 41" on page 342502227.

NOTE: If a faulty LRU is not reporting its software version and part number, check aircraft maintenance logs for last software version loaded and verify against the latest configuration.

(2) GDU 1050 PFD and GDU 1250A MFD

See "Figure 15" on page 34250163.

CAUTION: THE GDU 1050S USE A LENS COATED WITH A SPECIAL ANTI-REFLECTIVE COATING THAT IS VERY SENSITIVE TO SKIN OILS, WAXES AND ABRASIVE CLEANERS. CLEANERS CONTAINING AMMONIA WILL HARM THE ANTI-REFLECTIVE COATING. IT IS VERY IMPORTANT TO CLEAN THE LENS USING A CLEAN, LINT-FREE CLOTH AND AN EYEGLASS LENS CLEANER THAT IS SPECIFIED AS SAFE FOR ANTI-REFLECTIVE COATINGS.

(a) Description

Two Garmin GDU 1050 displays and one GDU 1250A display are installed in the instrument panel. The GDU 1050 units, 10.4 inch LCD displays with 1024x768 resolution, are configured as PFD1 and PFD2; the GDU 1250A unit, a 12 inch LCD display with 1080x800 resolution, is configured as a MFD. The PFD displays are located on either side of the MFD, with the stand-by instruments located to the left of the Pilot’s PFD (PFD1). A GMC 710 AFCS Controller is located in the center of the instrument panel, above or below the MFD. Additionally, a GMA 350c Audio Panel is located below the MFD, and a GCU 47X is installed in the pedestal. The GCU 47X provides the control interface for the MFD.

The GDU 1250A communicates with the GDU 1050 units and the GDL 69eA / 69A SXM through a high-speed data bus (HSDB) Ethernet connection. The GDU 1250A communicates with the GCU 47X via RS-232 digital interface.

The GDU 1050 units communicate with each other and the GIA 64(W) units through a High-Speed Data Bus (HSDB) Ethernet connection.

(b) Troubleshooting

See "Chart 9" on page 34250145,

"Chart 13" on page 34250164, and "Chart 14" on page 34250166.

PFD and MFD Displays
Figure 17

(c) Removal

1) Using a 3/32nd hex tool, rotate all four ¼-turn fasteners counter-clockwise until they reach their stops.

2) Carefully remove the display from the panel.

3) While supporting the display, disconnect the connector.

(d) Installation

1) Visually inspect the connector and pins for signs of damage. Repair any damage. While supporting the display, connect the connector to the rear of the unit.

2) Carefully insert the display into the panel cutout, ensuring that all 4 ¼-turn fasteners align with the corresponding holes.

3) Seat the display in the panel cutout. Do not use excessive force while inserting the display.

4) Once seated, rotate all four ¼-turn fasteners clockwise to lock the display to the panel.

(e) Return to Service

1) Original Display Reinstalled.

If the removed display(s) are re-installed in their original positions, no software or configuration loading is required. Continue to the "PFD/MFD Test" on page 342502181 procedure.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original PFD Displays Installed in Opposite Locations for Troubleshooting.

If the PFD1 and PFD2 are installed in opposite positions, no software or configuration loading is required. Continue to the"PFD/MFD Test" on page 342502181 procedure.

3) New, Repair, or Exchanged Display(s) Installed.

If a new, repaired or exchanged GDU 1050 or GDU 1250A is installed, the correct software and configuration files must be loaded to the unit. See "G1000 NXi Software/Configuration Procedure" on page 342502152 under Software Files. If ChartView or TAWS were previously installed, these must be reactivated. See Chartview and TAWS Unlock Procedure respectively. If any other options were previously installed, these must also be reactivated per "Optional Equipment Configuration" on page 342502158, then continue to the"PFD/MFD Test" on page 342502181.

CHART 13 (Sheet 1 of 2)
G1000 NXi - TROUBLESHOOTING - GDU

<table>
<tr><th>Symptom</th><th>Recommended Action</th></tr>
<tr><td>Display is blank</td><td rowspan="3">• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin.
• Cycle power.
✓ If GDU recovers, observe display for yellow text containing error information at the top of the screen. If message indicates software need to be re-loaded, then re-load software. Otherwise, replace the GDU.
• Use a bright light to verify LCD is active.
✓ Adjust avionics dimmer control full clockwise.
✓ Manually turn up backlight on the PFD and load configuration files to the GDU.
• Ensure slide lock is fully engaged with the locking tabs on the back of the unit.
✓ If slide lock is not fully engaged, remove connector and verify the locking tabs on the GDU are perpendicular to the connector. If necessary, straighten them before reseating connector.
• Ensure GDU is receiving power. If a circuit breaker is tripped, determine source of short before resetting breaker.
• Ensure circuit breakers have not failed and power wire connections are secure.
• Swap PFD1 and PFD2.
✓ If problem follows unit, replace the display. Please note the position it failed in (PFD1/2).
✓ If problem does not follow unit, troubleshoot aircraft wiring for fault.</td></tr>
<tr><td>Display resets</td></tr>
<tr><td>Display flickers</td></tr>
<tr><td>SD card is stuck in GDU</td><td>• DO NOT insert a screwdriver of any length into the card slot.
• DO NOT pry against the overlay.
• DO NOT force the SD Card out.
• Use a small screwdriver in the groove on the side of the exposed end of the card to help pull out the card.
• Push the card in further to release the card locking mechanism.
• Check SD Card for having more than one label. Two or more labels on the card will cause sticking.
✓ Remove all but one label.
• Ensure the SD card is from SanDisk. Use of other SD Cards is not recommended.
• If card was inserted with the label facing to the right, do not attempt to remove. Return the unit to Garmin for repair.</td></tr>
<tr><td>A button/knob/joystick does not appear to function</td><td>• Go to the GDU TEST page in configuration mode and verify button, knob, or joystick operates correctly by observing a change in color from red to green in the button/knob/joystick icon when the button/knob/joystick is pressed. If a button is stuck, the button icon will be green without pressing the button as soon as you turn to the GDU TEST page.
✓ If problem is verified, replace GDU.</td></tr>
</table>

CHART 13 (Sheet 2 of 2)
G1000 NXi - TROUBLESHOOTING - GDU

<table>
<tr><th>Symptom</th><th>Recommended Action</th></tr>
<tr><td>Terrain/Obstacle/Safetaxi does not display</td><td>• Ensure supplemental data cards are inserted correctly in the lower slots of all three GDU’s.
• Allow the system to verify the data on the cards for approximately five minutes after power-up.
• If a database does not activate, reload the problem database onto the SD Card or replace the card. Replacing all data cards in a set will keep database cycles the same and should prevent database mismatch errors.</td></tr>
<tr><td>Display will not track dimmer bus</td><td rowspan="2">• Reload GDU configuration files.
• If display is a PFD, swap PFD1 and PFD2 to see if problem remains with display.
✓ Replace display if condition remains with the same unit.
✓ If condition remains in original position after swapping displays, check GDU dimmer input to verify voltage is present.
• If display is the MFD, check dimmer input to verify voltage is present.
✓ Replace MFD if dimmer voltage is present.</td></tr>
<tr><td>Keyboard will not track dimmer bus</td></tr>
</table>

CHART 14 (Sheet 1 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

Failure Message	Cause	Solution
SW MISMATCH – GDU software version mismatch. Xtalk is off.	The system has found the PFDs and/or MFD software versions do not match.	• Load correct software version. See Software Loading procedure.
MANIFEST – PFD 1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in the specified PFD.	
MANIFEST – PFD 2 software mismatch. Communication Halted.		
MANIFEST – MFD1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in MFD.	
CNFG MODULE – PFD 1 configuration module is inoperative.	The PFD master configuration module has failed.	• Check master configuration module connector and wiring for damage inside the GDU connector backshell. ✓ Replace master configuration module wiring and pins. ✓ If problem persists, replace master configuration module. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

CHART 14 (Sheet 2 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

<table>
<tr><th>Failure Message</th><th>Cause</th><th>Solution</th></tr>
<tr><td>MFD1 CONFIG – MFD1 configuration error. Config service req’d.</td><td rowspan="3">A configuration mismatch has occurred between the display and the Master Configuration Module.</td><td rowspan="3">• Reload the display configuration files from SD Loader Card.
• Reload system configuration files by pressing the UPDT CFG softkey on the Configuration Upload Page in the PFD1 System Page Group to load configuration files into the configuration module.
✓ If message persists, check PFD1 config module wiring for faults and replace if necessary.
✓ If issue continues, replace PFD1 master configuration module.
✓ If problem persists, replace the display.

NOTE
New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.</td></tr>
<tr><td>PFD 1 CONFIG – PFD 1 configuration error. Config service req’d.</td></tr>
<tr><td>PFD 2 CONFIG – PFD 2 configuration error. Config service req’d.</td></tr>
</table>

CHART 14 (Sheet 3 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

<table>
<tr><th colspan="3">Database Message Advisories</th></tr>
<tr><th>Failure Message</th><th>Cause</th><th>Solution</th></tr>
<tr><td>MFD1 DB ERR – MFD1 navigation database error exists.

PFD1 DB ERR – PFD1 navigation database error exists.

PFD2 DB ERR – PFD2 navigation database error exists.</td><td>The MFD or specified PFD has encountered an error in the Jeppesen navigation database.</td><td>Reload navigation database into the display.

Contact Garmin Technical Support for assistance.</td></tr>
<tr><td>MFD1 DB ERR – MFD1 basemap database error exists.</td><td>The MFD has encountered an error in the basemap database.</td><td rowspan="8">Confirm supplemental data card is inserted fully in the bottom of the display.

Move the data card to the top slot of the display.

If the error clears, the problem is with the bottom slot. Insert and remove a SD card multiple times to clean the contacts. If the card still does not work in the bottom slot, leave the card in the top slot or replace the display.

Swap with a supplemental data card from another display in the same system.

For basemap only, if problem remains in the same GDU, contact Garmin Aviation Product Support to see if a Basemap file may be obtained to load into the display.

For all others, if problem remains in the same GDU, replace that GDU

If problem moves to the other display, replace the supplemental data card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.</td></tr>
<tr><td>PFD1 DB ERR – PFD1 basemap database error exists.

PFD2 DB ERR – PFD2 basemap database error exists.</td><td>The specified PFD has encountered an error in the basemap database.</td></tr>
<tr><td>MFD1 DB ERR – MFD1 terrain database error exists.</td><td>The MFD has encountered an error in the terrain database.</td></tr>
<tr><td>PFD1 DB ERR – PFD1 terrain database error exists.

PFD2 DB ERR – PFD2 terrain database error exists.</td><td>The specified PFD has encountered an error in the terrain database.</td></tr>
<tr><td>MFD1 DB ERR – MFD1 obstacle database error exists.</td><td>The MFD has encountered an error in the obstacle database.</td></tr>
<tr><td>PFD1 DB ERR – PFD1 obstacle database error exists.

PFD2 DB ERR – PFD2 obstacle database error exists.</td><td>The specified PFD has encountered an error in the obstacle database.</td></tr>
<tr><td>MFD1 DB ERR – MFD1 airport terrain database error exists.</td><td>The MFD has encountered an error in the airport terrain database.</td></tr>
<tr><td>PFD1 DB ERR – PFD1 airport terrain data base error exists.

PFD2 DB ERR – PFD2 airport terrain data base error exists.</td><td>The specified PFD has encountered an error in the airport terrain database.</td></tr>
</table>

CHART 14 (Sheet 4 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

Database Message Advisories (cont.)		
Failure Message	**Cause**	**Solution**
MFD1 DB ERR – MFD1 SafeTaxi database error exists.	The MFD has encountered an error in the SafeTaxi database.	Confirm supplemental data card is inserted fully in the bottom of the display. Move the data card to the top slot of the display. If the error clears, the problem is with the bottom slot. Insert and remove a SD card multiple times to clean the contacts. If the card still does not work in the bottom slot, leave the card in the top slot or replace the display. Swap with a supplemental data card from another display in the same system. If problem remains in the same GDU, replace that GDU. If problem moves to the other display, replace the supplemental data card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.
PFD1 DB ERR – PFD1 SafeTaxi data base error exists. PFD2 DB ERR – PFD2 SafeTaxi data base error exists.	The specified PFD has encountered an error in the SafeTaxi database.	
MFD1 DB ERR – MFD1 terrain database missing. PFD1 DB ERR – PFD1 terrain database emissing. PFD2 DB ERR – PFD2 terrain database missing.	The terrain database is present on another LRU, but is missing on the specified LRU.	Ensure data card is properly inserted. Swap with a supplemental data card from another display in the same system. If problem remains in the same GDU, replace that GDU. If problem moves with the data card, replace card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.

CHART 14 (Sheet 5 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

Database Message Advisories (cont.)		
Failure Message	**Cause**	**Solution**
MFD1 DB ERR – MFD1 obstacle database missing. PFD1 DB ERR – PFD1 obstacle database missing. PFD2 DB ERR – PFD2 obstacle database missing.	The obstacle database is present on another LRU, but is missing on the specified LRU.	Ensure data card is properly inserted. Swap with a supplemental data card from another display in the same system. If problem remains in the same GDU, replace that GDU. If problem moves with the data card, replace card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors.
MFD1 DB ERR – MFD1 airport terrain database is missing. PFD1 DB ERR – PFD1 airport terrain database is missing. PFD2 DB ERR – PFD2 airport terrain database is missing.	The airport terrain database is present on another LRU, but is missing on the specified LRU.	
MFD DB ERR – MFD ChartView database error.	The MFD has encountered an error in the ChartView database.	Confirm supplemental data card is inserted fully in the bottom of the display. Move the data card to the top slot of the display. If the error clears, the problem is with the bottom slot. Insert and remove a SD card multiple times to clean the contacts. If the card still does not work in the bottom slot, leave the card in the top slot or replace the display. Reload Chartview, FlightCharts, or Airport Directory, as appropriate. Replace the supplemental data card. You may need to replace all data cards as a set to keep the database cycles the same and prevent database mismatch errors. Replace MFD.
MFD DB ERR – MFD FliteCharts database error exists.	The MFD has encountered an error in the FliteCharts database.	
MFD1 DB ERR - MFD1 Airport Directory database error exists. PFD1 DB ERR - PFD1 Airport Directory database error exists.	The LRU has encountered an error in the Airport Directory Database.	

CHART 14 (Sheet 6 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

Database Message Advisories (cont.)

Failure Message	Cause	Solution
DB MISMATCH – Navigation database mismatch. Xtalk is off.	The G1000 NXi has found the Jeppesen navigation database in the PFDs and MFD do not match.	Load the same cycle of navigation database to all displays.
DB MISMATCH – Standby navigation database mismatch.	The G1000 NXi has found the standby Jeppesen navigation database in the PFDs and MFD do not match.	Load the same cycle of standby navigation database to all displays.
DB MISMATCH – Terrain database mismatch.	The PFD and/or MFD have different terrain database versions installed.	Confirm supplemental data card is inserted fully. Reload the database in each LRU and/or replace the supplemental datacard(s) in each LRU.
DB MISMATCH – Obstacle database mismatch.	The PFD and/or MFD have different obstacle database versions installed.	
DB MISMATCH – Airport Terrain database mismatch.	The PFD and/or MFD have different airport terrain database versions installed.	

Cooling Message Advisories

Failure Message	Cause	Solution
MFD1 COOLING – has poor cooling. Reducing power usage.	MFD1 has exceeded its operating temperature range.	Check cooling fan and wiring for proper operation. Replace cooling fan if unable to determine if operating correctly. If problem persists, replace the affected LRU. If problem continues contact Garmin Aviation Product Support for assistance.
PFD1 COOLING – has poor cooling. Reducing power usage.	PFD1 has exceeded it's operating temperature range.	
PFD2 COOLING – has poor cooling. Reducing power usage.	PFD2 has exceeded it's operating temperature range.	

CHART 14 (Sheet 7 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

Key Message Advisories		
Failure Message	**Cause**	**Solution**
MFD1 “key” KEYSTK – key is stuck.	The SYSTEM has determined a key is stuck on MFD1.	• Go to the GDU TEST page in configuration mode and verify key is stuck (if key is stuck the corresponding indicator will be green). • Exercise suspected stuck key and reset GDU TEST page to see if indicator remains green without pressing the key. ✓ If problem persists replace the display.
PFD 1 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 1.	
PFD 2 “key” KEYSTK – key is stuck.	The system has determined a key is stuck on the PFD 2.	

Miscellaneous Message Advisories		
Failure Message	**Cause**	**Solution**
XTALK ERROR – A flight display cross talk error has occurred.	A communication error has occurred between the MFD and/or PFDs.	• Ensure a database error has not occurred (identified in the ALERTS window on the PFD). ✓ If a database error has occurred, correct error before proceeding. • Check display Ethernet interconnect wiring. • Replace PFD1 with a known good unit, to verify location of problem: ✓ If problem persists, reinstall original PFD1 and replace PFD2. ✓ If problem persists, reinstall PFD2 and replace MFD.
DATA LOST – Pilot stored data lost. Recheck settings.	Pilot stored data has been lost.	• If the CLR key was held during a power cycle, disregard message. • Cycle power to PFD1: ✓ Ensure CLR key is not stuck on the GDU TEST page. ✓ If problem persists, replace PFD1.

CHART 14 (Sheet 8 of 8)
G1000 NXi - MESSAGE ADVISORIES - GDU

Miscellaneous Message Advisories		
Failure Message	**Cause**	**Solution**
MFD1 SERVICE – needs service. Return unit for repair.	The G1000 has determined MFD1 needs service.	• Ensure the MFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the MFD • Replace the MFD
PFD 1 SERVICE – needs service. Return unit for repair.	The G1000 has determined the specified PFD needs service.	• Ensure the PFD connector is fully seated and locked. • If the unit was started in a very dark environment the photocells may not have enough light to initially raise the CCFT level. Go to the GDU STATUS page in configuration mode, ensure CCFT CRNT 1 & 2 levels are above 50. ✓ If the CCFT levels are not above 50, apply light to the photocell and observe if the CCFT level rises. ✓ If the CCFT level rises, disregard the message. ✓ If the CCFT level does not rise, replace the PFD • Replace the PFD.
PFD 2 SERVICE – needs service. Return unit for repair.		
PFD 1 VOLTAGE – PFD 1 has low voltage. Reducing power usage.	The specified PFD supply voltage is low.	• Check input voltage to PFD. • If input voltage is ok, replace PFD..
PFD 2 VOLTAGE – PFD 2 has low voltage. Reducing power usage.		
MFD1 VOLTAGE – MDF1 has low voltage. Reducing power usage.	The MFD supply voltage is low.	• Check input voltage to MFD. • If input voltage is ok, replace MFD.

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(3) GCU 47X PFD/MFD Control Unit

See "Figure 1" on page 3425021 and "Figure 7" on page 34250212.

(a) Description

This pedestal mounted keypad provides MFD/PFD and radio tuning control through an RS-232 digital interface.

(b) Troubleshooting

See "Chart 15" on page 34250281.

(c) Removal

1) Turn each of the four locking sockets one quarter turn counterclockwise until they reach their stops using a 3/32” hex drive tool.

2) Pull unit up gently to remove unit from the panel.

3) Remove connector from the back of the unit by pushing on the slide lock tab on the side of the connector to release it and then pull connector away from unit. If unit connector uses thumb screws instead of a slide lock, they must be disconnected on both sides.

(d) Installation

1) Inspect wire harness connector for damaged pins before installing the new unit.

2) Attach connector to the back of the unit by first pressing the slide lock tab on the side of the connector, then align and mate the unit and wire harness connectors together, and then release the slide lock tab. Locking tabs must be engaged on both ends of the connector. If the unit connector uses thumb screws instead of a slide lock, they must be tightened on both sides securely.

3) Install and hold the unit flush with the instrument panel and ensure that locking stud alignment marks are in the vertical position.

4) Turn each of the four locking sockets one quarter turn clockwise using a 3/32” hex drive tool (this may require applying a small amount of forward pressure to engage the quarter turn sockets).

(e) Return to Service

1) Original GCU 47X Reinstalled

No software or configuration loading is required if the removed GCU 47X is re-installed. Continue to the "GCU 47X Test" on page 342502199.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GCU 47X Installed

If a new, repaired, or exchanged GCU 47X is installed, the correct software and configuration files must be loaded to the unit. See "G1000 NXi Software/Configuration Procedure" on page 342502152 and then continue to the "GCU 47X Test" on page 342502199.

CHART 15
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GCU 47X

Failure Messages	Cause	Solutions
GCU CNFG – GCU Config error. Config service req'd.	The G1000 NXi has detected a GCU 47X configuration	• Load GCU configuration files. ✓ Replace GCU. ✓ If problem persists, replace master configuration module(1), check config module wiring for faults and replace if necessary.
GCU FAIL – GCU is inoperative.	The G1000 NXi has detected a failure in the GCU 47X.	• Check wiring for faults per "Failed Path Messages." • Replace the GCU 47X.
MANIFEST – GCU software	The system has detected an incorrect software version loaded in the GCU 47X.	• Load the correct software version. See G1000 NXi Software/Configuration Procedures, below.
GCU KEYSTK – GCU [key name] key is stuck.	A key is stuck on the GCU bezel.	Exercise the key to free it. If the problem persists, replace the GCU 47X.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

(4) GMA 350c Audio Panel

See "Figure 18".

(a) Description

The Garmin GMA 350c Audio Panel integrates NAV/COM digital audio, intercom system and marker beacon controls. The GMA 350c panel provides control of all cockpit intercom/ mic systems as well as NAV/COM/ILS audio. Power to the audio panel is provided by the battery switch or by applying external power. The GMA 350c unit interfaces with the marker beacon antenna as well as mic and phone jacks and oxygen mask mic.

(b) Troubleshooting

See "Chart 16" on page 34250283 and "Chart 17" on page 34250284.

(c) Removal

1) Using a 3/32nd hex tool, turn the hex nut counter-clockwise until the GMA 350c is unlocked from its location.

2) Carefully remove the GMA 350c from its rack.

(d) Installation

1) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.

2) Gently insert the GMA 350c into the rack until the locking tab engages the rack.

3) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not overtighten the nut.

(e) Return to Service

1) Original GMA 350c Reinstalled.

No software/configuration loading or testing is required if the removed GMA 350c is reinstalled, continue to the "GMA 350c Test" on page 342502181 under LRU Test Procedures.

NOTE: This does not include units that were returned for repair (see below) as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GMA 350c Installed.

If a new, repaired, or exchanged GMA 350c is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 NXi Software/Configuration Procedure" on page 342502152, then continue to the"GMA 350c Test" on page 342502181.

GMA 350c Audio Panel
Figure 18

CHART 16
G1000 NXi - TROUBLESHOOTING - GMA

Symptom/Failure Message	Recommended Action
Noise in Audio	Most often the cause of the noise is external to the GMA. Try the following to locate the source of the noise before replacing the GMA: • Try a different pair of headsets. Noise cancelling headsets may pick up and/or generate more noise than standard headsets from their own circuitry. • Check for noise with the engine turned off. ✓ If the noise is present only when the engine is running, check the generator and/or ignition system as possible sources of noise (see applicable airframe specific maintenance manual). • Check for noise as all electrical equipment is turned on and off (strobes, other radios, etc.). ✓ If the noise is identified from one electrical system or component refer to the applicable airframe specific maintenance manual. • Ensure the NAV/COM squelch is not open. • Ensure the ADF and DME audio is not active. • Ensure the marker beacon audio is not active. • Ensure the ICS squelch is not open. ✓ Master squelch level can be adjusted on the GMA CONFIGURATION page for higher noise environments. • Replace unit only after all possible external sources of noise are eliminated.
Buttons Do Not Work.	• Some buttons are disabled in the GMA CONFIGURATION page by default. This is to remove potential sources of audio noise for inputs that are not used. If in doubt as to which buttons should be disabled, reload GMA config files and other config files for optional equipment installed in the aircraft (i.e. ADF, HF, etc.) from the loader card.
Speaker Cuts Out	• Reduce volume level of the item that caused the speaker to cut out when turned up. A speaker protection circuit disables the speaker output if the volume is too high. If the volume is not sufficient, replace aircraft cabin speaker.
Mic Audio Heard in Speaker	• Reduce ICS Volume.

CHART 17 (Sheet 1 of 2)
G1000 NXi - MESSAGE ADVISORIES - GMA

Failure Message	Cause	Solution
MANIFEST – GMA1 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GMA #1.	Load correct software version.
MANIFEST – GMA2 software mismatch. Communication Halted.	The system has detected an incorrect software version loaded in GMA #2.	Load correct software version.
GMA1 SERVICE – GMA1 needs service. Return unit for repair.	The G1000 NXi has determined GMA#1 needs service.	Replace GMA #1.
GMA2 SERVICE – GMA2 needs service. Return unit for repair.	The G1000 NXi has determined GMA#2 needs service.	Replace GMA #2.
GMA1 FAIL – GMA 1 is inoperative.	The G1000 NXi has detected a failure in GMA#1.	Ensure GMA #1, both GIAs and all GDUs are receiving power. Troubleshoot for a failed datapath per “Failed Path Messages.” Replace GMA #1.
GMA2 FAIL – GMA 2 is inoperative.	The G1000 NXi has detected a failure in GMA#2.	Ensure GMA #2, both GIAs and all GDUs are receiving power. Troubleshoot for a failed datapath per “Failed Path Messages.” Replace GMA #2.
GMA1 CONFIG – GMA1 configuration error. Config service req’d.	The G1000 NXi has detected a GMA #1 configuration mismatch.	Reload affected GMA Configuration files including optional equipment configuration files that list that GMA in the PRODUCT box on the System Upload page. Replace affected GMA. If problem persists, replace master configuration module, check config module wiring for faults and replace if necessary.
GMA2 CONFIG – GMA2 configuration error. Config service req’d.	The G1000 NXi has detected a GMA #2 configuration mismatch.	

NOTE: New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

CHART 16 (Sheet 2 of 2)
G1000 NXi - MESSAGE ADVISORIES - GMA

Failure Message	Cause	Solution
GMA XTALK – GMA crosstalk error has occurred.	The G1000 NXi has detected an error in the communication between GMA #1 and GMA #2.	Ensure both units are receiving power. Ensure both units are configured. Check interconnect wiring and connector pins for faults. Replace GMA #1 with a known good unit. If problem persists, reinstall original GMA #1 and replace GMA #2.
BACKUP PATH – Audio panel using backup data path.	The GMA is using a backup RS-232 data path.	Troubleshoot for a failed per “Failed Path Messages.”

(5) GIA 64(W) Integrated Avionics Unit

See "Figure 19".

In S/N's 4636716, 4636720-4636759, GIA 64 is installed. In S/N's 4636760 and up, GIA 64W is installed.

(a) Description

Two Garmin GIA 64(W) Integrated Avionics Units (IAUs) contain the VHF COM/NAV receivers, WAAS GPS receiver, Flight Director, and system integration microprocessors. The GIAs also serve as a communication interface to all other G1000 NXi LRUs in the system. Each GIA communicates directly with the on-side GDU 1050 display using a HSDB Ethernet connection. Both GIAs are located directly behind the PFD1 and PFD2, installed into their respective LRU racks. Power is provided to GIA1 by the main bus or the emergency bus. GIA2 is powered by #2 avionics bus.

GIA 64(W) Integrated Avionics Unit
Figure 19

1) Both GIAs interface to the following equipment:

- VOR/LOC/Glideslope Antenna System
- VHF COM #1 & #2 Antennas
- GA 35 GPS/WAAS Antennas
- GMA 350c
- GEA 71B,
- GDU 1050, #1 & #2
- GSA 8X Servo Actuators (all)
- GRS 79, #1 & #2

2) GIA #1 interfaces to the following additional equipment:

- GDC 72 #1
- GTX 335R #1 (GTX 33 ES or GTX 345R Optional) (S/N's 4636716, 4636720-4636759)
  or
- GTX 345R #1 (GTX 345DR Optional) (S/N's 4636760 and up)
- GTS 825 TAS (if Installed)

3) GIA #2 interfaces to the following additional equipment:

- GDU 1250A
- GDC 72 #2
- GTX 335R #2 (GTX 33 ES or GTX 345R Optional) (S/N's 4636716, 4636720-4636759) or
- GTX 345R (GTX 335R or GTX 345DR optional) (S/N's 4636760 and up)
- ADF (if installed)
- DME (if Installed)
- KTA 810 TAS (if Installed)
- Stormscope (if installed)

NOTE: The GIA 64(W) is compatible only with the antennas listed in the GIA 64(W) Installation Manual, Garmin P/N 190-01912-00.

(b) Troubleshooting

See "Chart 18" on page 34250288 and "Chart 19" on page 34250290.

(c) Removal

1) Remove the appropriate PFD as described above.
2) Unlock the GIA handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GIA. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GIA into its rack. The handle should engage the dogleg track.
3) Press down on the GIA hand to lock the unit into the rack.
4) Lock the handle to the GIA body using the Phillips screw.
5) Reinstall the MFD as described above.

(e) Return to Service

1) Original GIA(s) Reinstalled

No software or configuration loading is required if the removed GIA is re-installed in its original position (GIA1 and GIA2 in their original racks).

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GIA(s) Swapped for Troubleshooting

No software loading is required if the originally installed GIA units are re-installed in opposite positions (GIA1 and GIA2 in opposite unit racks). However, configuration loading is required.

See "G1000 NXi Software/Configuration Procedure" on page 342502152, and then continue to the "GIA 64(W) Test" on page 342502184.

3) New, Repaired, or Exchanged GIA(s) Installed

If a new, repaired, or exchanged GIA is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 NXi Software/Configuration Procedure" on page 342502152, then continue to the "GIA 64(W) Test" on page 342502184.

CHART 18 (Sheet 1 of 2)
G1000 NXi - TROUBLESHOOTING - GIA 64(W)

COM	
Symptom	**Recommended Action**
Weak COM transmit power	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
Weak COM receiver	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check COM antenna and cabling for faults.
No COM sidetone	• Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem persists, replace GMA1 with a known good unit. ✓ If problem persists, reinstall original GMA1 and replace GMA2.

NAV	
Symptom	**Recommended Action**
Weak NAV receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

Glideslope	
Symptom	**Recommended Action**
Weak G/S receiver	• Set up a NAV/COM Ramp Test Set to radiate a test signal. • Switch GIA1 and GIA2, to verify location of problem: ✓ If problem follows unit, replace GIA. ✓ If problem does not follow unit, check NAV antenna, coupler, and cabling for faults.

CHART 17 (Sheet 2 of 2)
G1000 NXi - TROUBLESHOOTING - GIA 64(W)

GPS	
Symptom	**Recommended Action**
Will not acquire satellites.	Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.). Check the aircraft records for the model and serial number of the GPS antenna. If Garmin GA 35 antennas below S/N 60000 are installed, one of the antennas may be the source of interference. Replace both GPS antennas to correct. Check date and time on Date/Time Setup Page. If date and time are incorrect, enter the correct date and time. Swap GIA1 and GIA2, reconfigure both GIA's to their new locations, to verify location of problem. If problem follows unit, clear the GPS almanac by performing the following steps: Using the PFD in config mode, go to the GIA RS-232/ARINC 429 Config Page. At the top of the screen, select the GIA that cannot acquire satellites (GIA1 or GIA2) and press the ENT key. Press the "CLR NV" softkey at the bottom of the screen. Select "OK" in the "Clear GIA nonvolatile memory?" pop-up window. Reload GIA Audio and Config files from a loader card. Be sure to reload the config files for any optional equipment installed on the aircraft that require the GIA config to be updated. Cycle power on the system and allow it to restart in normal mode. Place the aircraft outside and allow 15-30 minutes for the GPS to acquire a position and download a new almanac. If clearing nonvolatile memory is unsuccessful and the GPS still cannot acquire a position, replace the GIA. Check GPS antenna and cabling.

CHART 19 (Sheet 1 of 7)
G1000 NXi - MESSAGE ADVISORIES - GIA 64(W)

COM		
Failure Message	**Cause**	**Solutions**
COM1 SERVICE – COM1 needs service. Return unit for repair.	The system has determined COM1 needs service.	• Replace GIA1.
COM2 SERVICE – COM2 needs service. Return unit for repair.	The system has determined COM2 needs service.	• Replace GIA2.
COM1 PTT – COM1 push-to-talk key is stuck.	The COM1 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA1/GMA #1 interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped; see Software/Configuration above): ✓ If problem follows the unit, replace GIA1. ✓ If problem persists replace defective GMA #1.
COM2 PTT – COM2 push-to-talk key is stuck.	The COM2 external push-to-talk (PTT) switch is stuck in the enabled (or “pressed”) state.	• Press the push-to-talk switch(s) again to cycle its operation. • Check push-to-talk switch(s) and wiring. • Check GIA2/GMA (#2, if installed) interconnect. • Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/Configuration above): ✓ If problem follows the unit, replace GIA2. ✓ If problem persists replace defective GMA (#2, if installed).

CHART 18 (Sheet 2 of 7)
G1000 NXi - MESSAGE ADVISORIES - GIA 64(W)

<table>
<tr><th colspan="3">COM</th></tr>
<tr><th>Failure Message</th><th>Cause</th><th>Solutions</th></tr>
<tr><td>COM1 RMT XFR – COM1 remote transfer key is stuck.</td><td>The COM1 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td><td>Press the COM1 external remote transfer switch again to cycle its operation.
Check COM1 external remote transfer switch and wiring.
Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above):
✓ If problem follows the unit, replace GIA1.
✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.</td></tr>
<tr><td>COM2 RMT XFR – COM2 remote transfer key is stuck.</td><td>The COM2 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td><td>Press the COM2 external remote transfer switch again to cycle its operation.
Check COM2 external remote transfer switch and wiring.
Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above):
✓ If problem follows the unit, replace GIA2.
✓ If problem persists, continue to troubleshoot remote transfer switch & wiring.</td></tr>
<tr><td>COM1 TEMP – COM1 over temp. Reducing transmitter power.</td><td rowspan="2">The specified COM transceiver is reporting a high temperature condition and is reducing transmit power to prevent damage.</td><td rowspan="2">• Check fan, wiring and air tubing for proper operation (if applicable).
• Replace cooling fan if unable to determine if operating correctly.
• Replace GIA.
• If problem persists contact Garmin Aviation Product Support for assistance.</td></tr>
<tr><td>COM2 TEMP – COM2 over temp. Reducing transmitter power.</td></tr>
</table>

CHART 18 (Sheet 3 of 7)
G1000 NXi - MESSAGE ADVISORIES - GIA 64(W)

<table>
<tr><th colspan="3">NAV</th></tr>
<tr><th>Failure Message</th><th>Cause</th><th>Solution</th></tr>
<tr><td>NAV1 SERVICE – NAV1 needs service. Return unit for repair.</td><td>The system has detected a failure in NAV1 receiver.</td><td>Replace GIA1.</td></tr>
<tr><td>NAV2 SERVICE – NAV2 needs service. Return unit for repair.</td><td>The system has detected a failure in NAV2 receiver.</td><td>Replace GIA2.</td></tr>
<tr><td>NAV1 RMT XFR – NAV1 remote transfer key is stuck.</td><td>The NAV1 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td><td rowspan="2">Press the affected NAV external remote transfer switch again to cycle its operation.

Check affected NAV remote transfer switch and wiring.

Switch GIA1 and GIA2, to identify whether the unit or connectors/wiring is at fault (Both GIAs must be configured when swapped, see Software/ Configuration above):

If problem follows unit, replace affected GIA.

If problem persists, it’s in the remote transfer switch and/or wiring.</td></tr>
<tr><td>NAV2 RMT XFR – NAV2 remote transfer key is stuck.</td><td>The NAV2 external remote transfer switch is stuck in the enabled (or “pressed”) state.</td></tr>
<tr><td>NAV1 MANIFEST – NAV1 software mismatch. Communication halted.</td><td>The system has detected an incorrect software version loaded in GIA1.</td><td>Load the correct software. See G1000 NXi Software Configuration Procedure, below.</td></tr>
<tr><td>NAV2 MANIFEST – NAV2 software mismatch. Communication halted.</td><td>The system has detected an incorrect software version loaded in GIA2.</td><td>Load the correct software. See G1000 NXi Software Configuration Procedure, below.</td></tr>
</table>

CHART 18 (Sheet 4 of 7)
G1000 NXi - MESSAGE ADVISORIES - GIA 64(W)

Glideslope		
Failure Message	**Cause**	**Solution**
G/S1 SERVICE – G/S1 needs service. Return unit for repair.	The system has detected a failure in G/S1 receiver.	Replace GIA1.
G/S2 SERVICE – G/S2 needs service. Return unit for repair.	The system has detected a failure in G/S2 receiver.	Replace GIA2.
G/S1 FAIL – G/S1 is inoperative.	The system has detected a failure in G/S1 receiver.	Switch GIA1 and GIA2 to verify location of problem: If problem follows the unit, replace affected GIA. If problem does not follow unit, check affected G/S antenna and cabling.
G/S2 FAIL – G/S2 is inoperative.	The system has detected a failure in G/S2 receiver.	

GPS		
Failure Message	**Cause**	**Solution**
NOTE: Before troubleshooting, ensure that no cell phones or devices using cell phone technology are turned on, even in a monitoring state, in the cabin.		
GPS1 SERVICE – GPS1 needs service. Return unit for repair.	The system has detected a failure in GPS1 receiver.	Replace GIA1.
GPS2 SERVICE – GPS2 needs service. Return unit for repair.	The system has detected a failure in GPS2 receiver.	Replace GIA2.
GPS1 FAIL – GPS1 is inoperative.	The system has detected a failure in GPS1 receiver.	Switch GIA1 and GIA2, to verify location of problem: If problem follows the unit, replace affected GIA. If problem does not follow the unit, check affected GPS antenna and cabling.
GPS2 FAIL – GPS2 is inoperative.	The system has detected a failure in GPS2 receiver.	

CHART 18 (Sheet 5 of 7)
G1000 NXi - MESSAGE ADVISORIES - GIA 64(W)

<table>
<tr><th colspan="3">GPS (cont.)</th></tr>
<tr><th>Failure Message</th><th>Cause</th><th>Solution</th></tr>
<tr><td>LOI – GPS integrity lost. Crosscheck with other NAVS.</td><td>If the primary receiver is a WAAS sensor, the alert indicates that GPS position data has timed out.</td><td rowspan="4">• Verify the area the aircraft was traveling through did not have loss of GPS coverage. FAA NOTAMs may be issued for periods of outages, or the US Coast Guard website http://www.navcen.uscg.gov/ will have notices posted.
• Using the MFD AUX – GPS Status page, verify the signal strength bars are not erratic. If so, this indicates outside interference is affecting the GPS receivers. Find and remove the source of interference (i.e. cell phones, FBO datalink antennas, etc.).
• If GPS receivers can not aquire a position lock, troubleshoot per the “Will not acquire satellites” section.
• Check GPS antenna and cabling.</td></tr>
<tr><td>GPS NAV LOST – Loss of GPS navigation. Insufficient satellites.</td><td>There is no GPS position fix available or the system is in dead reckoning mode.</td></tr>
<tr><td>GPS NAV LOST – Loss of GPS navigation. Position error.</td><td>The G1000 has detected an internal position warning has occurred.</td></tr>
<tr><td>GPS NAV LOST – Loss of GPS navigation. GPS fail.</td><td>The G1000 has detected a GPS engine failure.</td></tr>
</table>

Cooling		
Failure Message	**Cause**	**Solution**
GIA1 COOLING – GIA1 temperature too low.	GIA1 operating temperature is too low.	• Allow unit to warm up.
GIA2 COOLING – GIA2 temperature too low.	GIA2 operating temperature is too low.	• Allow unit to warm up.
GIA1 COOLING – GIA1 over temperature.	GIA1 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA1. • If problem persists contact Garmin Aviation Product Support for assistance.
GIA2 COOLING – GIA2 over temperature.	GIA2 has exceeded its operating temperature range.	• Check fan, wiring and air tubing for proper operation (if applicable). • Replace cooling fan if unable to determine if operating correctly. • Replace GIA2. • If problem persists contact Garmin Aviation Product Support for assistance.

CHART 18 (Sheet 6 of 7)
G1000 NXi - MESSAGE ADVISORIES - GIA 64(W)

Configuration		
Failure Message	**Cause**	**Solution**
MANIFEST – GIA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA1.	Load the correct G1000 NXi system software. See G1000 NXi Software/Configuration Procedure, below.
MANIFEST – GIA2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in GIA2.	
MANIFEST – GFC software mismatch. Communication halted.	Incorrect servo software installed or gain settings are incorrect.	
COM1 CONFIG – COM1 configuration error. COM2 CONFIG – COM2 configuration error.	COM1 and/or COM2 configuration settings do not match backup configuration memory.	
COM1 MANIFEST – COM1 software mismatch. COM2 MANIFEST – COM2 software mismatch.	COM1 and/or COM2 software mismatch, communication halted.	
GIA1 CONFIG – GIA1 audio config error. Config service req'd. GIA2 CONFIG – GIA2 audio config error. Config service req'd.	The GIA's audio configuration files are incorrect or missing.	Reload GIA Audio software and configuration files. Replace master configuration module[(1)], check config module harness for faults and replace if necessary.
GIA1 CONFIG – GIA1 configuration error. Config service req'd. GIA2 CONFIG – GIA2 configuration error. Config service req'd.	The system has detected a GIA configuration mismatch. If GIAs are not properly configured after being swapped/replaced, this message appears.	Load the configuration files for that GIA. See G1000 NXi Software/ Configuration Procedure, below. If problem persists, replace master configuration module[(1)], check config module harness for faults and replace if necessary.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

CHART 18 (Sheet 7 of 7)
G1000 NXi - MESSAGE ADVISORIES - GIA 64(W)

Configuration (cont.)		
Failure Message	**Cause**	**Solution**
HW MISMATCH – GIA hardware mismatch, GIA1 communication halted. HW MISMATCH – GIA hardware mismatch, GIA2 communication halted.	The G1000 NXi has detected a non-WAAS GIA.	Replace GIA with a WAAS unit.
GIA1 SERVICE – GIA1 needs service. Return unit for repair. GIA2 SERVICE – GIA2 needs service. Return unit for repair.	The G1000 NXi has detected a failure in the specified GIA.	Replace suspect GIA.

(6) GAE 43 Cabin Altitude Encoder

See "Figure 20".

(a) Description

In PA-46-350Ps the GAE 43 Altitude Encoder is installed to function as the cabin pressure transducer. It generates cabin altitude and rate information that's collected by the GIA 64s and normally displayed on the MFD. It's mounted to the underside of the equipment shelf located forward of the MFD

(b) Removal

1) Disconnect the two avionic harness connectors.

2) Unscrew the thumbscrew and slide the unit down and aft to free it from its mounting tray.

(c) Installation

1) Position the unit below and aft of its mounting tray.

2) Engage the lips of the mounting tray and slide the unit forward and up until the thumbscrew can be engaged.

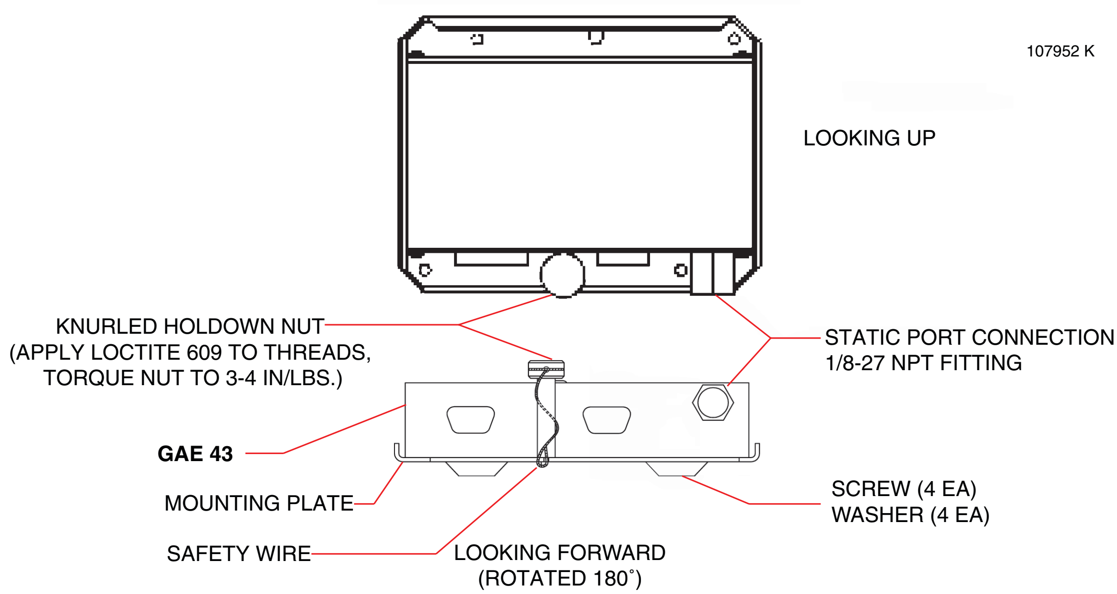
3) Tight thumbscrew to secure unit.

4) Connect the two avionic harness connectors.

(d) Calibration

See "GAE 43 Calibration" on page 342502202 under LRU Test Procedures.

NOTE: Calibrate a new unit before installation.



GAE 43 Altitude Encoder
Figure 20

(7) GEA 71B Engine/Airframe Unit

See "Figure 21" on page 34250298.

(a) Description

The Garmin GEA 71B Engine/Airframe Unit provides engine/airframe data to the G1000 NXi system. Data received from transducers/sensors is processed and sent to GIA 64s (via RS-485 digital interface), and subsequently to the GDU 1250A MFD. Engine parameters are normally displayed on the MFD. In the event of a MFD failure, all engine instruments can be displayed on the PFDs. The GEA is located behind the instrument panel and is mounted in a vertical orientation. Power is received from the main bus or the Emergency bus.

GEA 71B Engine/Airframe Unit
Figure 21

The GEA interfaces to the following:

- Oil Pressure Sensor
- Oil Temperature Sensor
- Fuel Flow Sensor
- Fuel Filter Sensor
- Fuel Quantity Sendors (left and right).
- Propeller Speed Sensor
- Pitot Heat System
- Flap Position potentiometer.
- Vacuum Transducer.
- Door Ajar switch

(b) Troubleshooting

See "Chart 20" on page 34250299.

(c) Removal

1) Remove PFD2 per "Removal" on page 34250268.
2) Unlock the GEA 71B handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GEA 71B.
4) Gently remove the GEA 71B from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GEA 71B into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GEA 71B body using the Phillips screw.
5) Reinstall PFD2 per "Installation" on page 34250268.

(e) Return to Service

1) Original GEA 71B Reinstalled

No software or configuration loading is required if the removed GEA 71B is reinstalled. Continue to the return-to-service checks.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GEA 71B Installed

If a new, repaired, or exchanged GEA 71B is installed, correct software and configuration files must be loaded to the unit.

See "G1000 NXi Software/Configuration Procedure" on page 342502152, and then continue to the "GEA 71B Test" on page 342502186.

CHART 20
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GEA

Failure Message	Cause	Solution
MANIFEST – GEA1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GEA 71B.	Load the correct software version. See G1000 NXi Software/Configuration Procedure, below.
GEA1 CONFIG – GEA1 configuration error. Config service req'd.	The system has detected a configuration mismatch in the GEA 71B.	Load GEA configuration files. See G1000 NXi Software/ Configuration Procedure, below. If problem persists, replace master configuration module(1), check config module harness for faults and replace if necessary.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

(8) GEA 71B Backshell Thermocouple

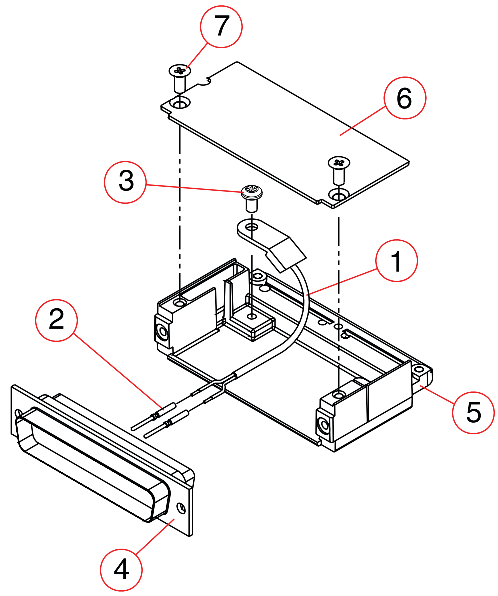
See "Figure 22".

(a) Removal

1) Remove GEA 71B "Removal" on page 34250298.
2) Remove GEA connector backplate.
3) Remove connector J701 (5) from the backplate.
4) Remove cover (6) from the backshell.
5) Unscrew thermocouple from boss on backshell. Extract the thermocouple pins from the connector.

(b) Installation

1) Crimp pins (2) onto each of the thermocouple wires (1). Ensure that pre-stripped wire length is 1/8” prior to crimping.
2) Insert newly crimped pins and wires into the appropriate connector housing location (4) as specified in the electrical schematic.
3) Place thermocouple body (1) onto the backshell boss (5). Place the thermocouple as shown in "Figure 22" so that the wires exit towards the bottom of the backshell.
4) Fasten thermocouple tightly to backshell using the provided screw (3).
5) Fasten cover (6) to backshell using the provided screws (7).
6) Reinstall GEA 71B per "Installation" on page 34250298.



GEA 71B Backshell Thermocouple
Figure 22

(9) GTX 3X5R Mode S/ES Transponder

See "Figure 23". In S/N's 4636720–4636775, the GTX 335R is standard, a GTX 345R is optional. In S/N's 4636776 and up, the GTX 345R is standard, a GTX 335R and GTX 345DR is optional.

(a) Description

The unit is mounted directly behind the MFD on the center avionics shelf. Power is received from the Essential Bus. Similarly to the GEA 71B, the GTX 3X5R sends data via RS-232 directly to a GIA 64(W). Information is then sent to the PFD, where the pilot can control the transponder. The GTX 3X5R is connected to both GIAs for redundant communications. The GTX 3X5R interfaces with the transponder antenna.

Each provides Mode A, C, and S altitude and position reporting information for the G1000 NXi system as well as TIS-A and ADS-B OUT (1090 ES). Using precise GPS-referenced positioning information, the extended squitter technology enables transponders to automatically transmit traffic surveillance data – including aircraft flight ID, position, altitude, velocity, climb/descent, and heading information.

GTX 3X5R Transponder
Figure 23

When the GTX 345R or 345DR is installed, ADS-B IN (UAT, TIS-B, and FIS-B) capability is added. The unit receives and displays ADS-B transmissions from other ADS-B OUT equipped aircraft, ADS-R, and TIS-B information from ground stations. In addition, FIS-B information from UAT ground stations in the United States is received and available for display. This includes: NOTAM (includes TFRs), AIRMET, SIGMET, SUA, METAR, TAFs, PIREP, Winds/Temps Aloft, Regional NEXRAD, CONUS NEXRAD.

(b) Troubleshooting

See "Chart 21" on page 342502103.

(c) Each Twenty-four (24) Calendar Months

The integrated transponder/altitude reporting system must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) §§ 91.411 and 91.413, every 24 calendar months, or any time the transponder is removed. This test requires the use of a Mode S ramp generator. Specific instructions for operating the ramp tester are contained in the applicable operator's manual. Refer to Title 14 CFR Part 43.

(d) Removal

1) Remove the MFD per "Removal" on page 34250268.
2) Unlock the GTX 3X5R handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GTX 3X5R. Gently remove the unit from the rack.

(e) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

2) Gently insert the GTX 3X5R into its rack. The handle should engage the dogleg track.

3) Press down on the GTX 3X5R handle to lock the unit into the rack.

4) Lock the handle to the GTX 3X5R body using the Phillips screw.

5) Reinstall the MFD per "Installation" on page 34250268.

(f) Return to Service

1) Original GTX 3X5R is Reinstalled

No software or configuration loading is required if the removed GTX 3X5R is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to "GTX 3X5R Test" on page 342502210.

2) New, Repaired or Exchanged GTX 3X5R is Installed

If a new, repaired or exchanged GTX 3X5R is installed, the correct software and configuration files must be loaded to the unit. See "GTX 3X5R Mode S/ES Transponder" on page 342502101.

NOTE: If performing a manual Complete Software Load, after loading software and configuration files continue to "Aircraft Registration Number Entry (Transponder Configuration)" on page 342502161.

Then, continue to "GTX 3X5R Test" on page 342502210.

CHART 21
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GTX 3X5R

Failure Message	Cause	Solutions
MANIFEST – GTX1 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GTX 3X5R.	• Reload software. See Software Load / Configuration Procedure.
MANIFEST – GTX2 software mismatch. Communication halted.		
XPDR1 CONFIG – XPDR1 configuration error. Config service req'd.	The system has detected a configuration mismatch for the specified GTX 3X5R.	• Perform a SET>ACTV configuration reset on the GTX Config page and verify the aircraft registration is present. • If error is still present, reload config files from a loader card. ✓ If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.
XPDR2 CONFIG – XPDR2 configuration error. Config service req'd.		
XPDR1 SRVC – XPDR1 needs service. Return unit for repair.	The G1000 NXi has detected a failure in the specified GTX 3X5R.	• Replace GTX 3X5R.
XPDR2 SRVC – XPDR2 needs service. Return unit for repair.		
XPDR1 FAIL – XPDR 1 is inoperative.	The specified GTX 3X5R is not responding.	• Check wiring between GIA's and GTX. • Replace GTX 3X5R.
XPDR2 FAIL – XPDR 2 is inoperative.		

(10) GDC 72 Digital Air Computer

See "Figure 24".

(a) Description

The GDC 72 computers compile information from the pitot/static system and the GTP 59 outside air temperature (OAT) probe to digital air data computations to the G1000 NXi system. The GDC 72 communicates with the GIA 64, GDU 1050, and GRS 79 using ARINC 429 digital interface. The units are mounted behind the MFD. Power is provided to #1 GDC 72 unit from the main bus or the emergency bus. The #2 GDC 72 is powered by the #2 avionics bus.

GDC 72 Digital Air Computer
Figure 24

(b) Troubleshooting

See "Chart 22" on page 342502105.

(c) Removal

1) Remove the MFD per "Removal" on page 34250268.

2) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.

3) Loosen each thumbscrew on the hold-down clamp and remove the clamp.

4) Carefully remove the unit from its mount.

(d) Installation

1) Place the unit in the mounting tray.

2) Position the locking clamp and fasten using the thumbscrews.

3) Connect the pitot/static plumbing.

4) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.

5) Reinstall the MFD per "Installation" on page 34250268

(e) Return to Service

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, Test, in 34-10-00.

1) Original GDC 72 is Reinstalled

No software or configuration loading is required if the removed GDC 72 is re-installed. Continue to "GDC 72 Air Data Computer Tests" on page 342502188.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GDC 72 Installed in Opposite Locations for Troubleshooting

No software loading is required if the original GDC1 and GDC2 are installed in opposite locations. Continue to "GDC 72 Air Data Computer Tests" on page 342502188.

3) New, Repaired, or Exchanged GDC 72 is Installed

If a new, repaired, or exchanged GDC 72 is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 NXi Software/Configuration Procedure" on page 342502152, and then continue to "GDC 72 Air Data Computer Tests" on page 342502188.

4) New GDC 72 Configuration Module is Installed

The correct configuration files must be loaded if the GDC 72 configuration module has been replaced.

See "G1000 NXi Software/Configuration Procedure" on page 342502152, then continue to "GDC 72 Air Data Computer Tests" on page 342502188.

CHART 22
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GDC 72

Symptom	Recommended Action
Altitude is different than standby altimeter.	Perform a pitot/static check per Pitot Static System Test in 34-10-00. Allow the GDC to warm up for fifteen minutes before checking accuracy. Determine which instrument is outside limits and recalibrate or replace. Note: Both units may individually be in spec but show a difference in altitude. Do not return the GDC to Garmin for service if not outside limits.
GDC Config file does not load.	Replace GDC config module. If problem persists, replace GDC config module wire harness.

Failure Message	Cause	Solution
MANIFEST – GDC1 software mismatch Communication halted. MANIFEST – GDC2 software mismatch Communication halted.	The system has detected an incorrect software version loaded in the specified GDC.	Load correct software version. See G1000 NXi Software/ Configuration Procedure, below.

(11) GTP 59 OAT Probe

See "Figure 25".

(a) Description

The Garmin GTP 59 OAT Probe provides the GDC 72 with air temperature data. The OAT probes are mounted to an access cover on the left wing.

GTP 59 OAT Probe
Figure 25

(b) Troubleshooting

See "Chart 5" on page 34250235.

(c) Removal

1) Remove the access cover.
2) Disconnect the connector.
3) Use an open-end wrench to hold the probe in place on the inside of the access cover. On the outside of the access cover, loosen the GTP 59 Housing and remove the GTP 59.

(d) Installation

1) Place the GTP 59 grounding strap O-ring connector over the OAT probe end and insert the probe through the access cover from the inside side. Secure with housing and washer on the outside of the access cover. Use an open-end wrench to hold the GTP 59 on the inside and torque the housing on the outside to 50 ± 5 in. lbs.
2) Reconnect the connector to the airplanes wiring harness.
3) Install the access cover.

(12) GRS 79 Attitude and Heading Reference System

See "Figure 26".

(a) Description

The Garmin GRS 79 AHRS units provide attitude and heading information to the G1000 NXi system. The units, mounted behind the pilots seat, contain advanced tilt sensors, accelerometers, and rate sensors. The unit interfaces with the GDC 72 and GMU 44(B) Magnetometer and utilizes GPS signals from the GIA 64(W)s. Actual attitude and heading information is sent using ARINC 429 digital interface to both GDU 1050s and GIA 64(W)s. Power is provided to #1 GRS 79 unit from the main bus or the emergency bus. The #2 GDC 72 is powered by the #2 avionics bus. The GRS 79 interfaces with and provides power to the GMU 44(B) Magnetometer. The GRS 79 supplies attitude and heading information directly to the PFDs, MFD, and GIAs.

GRS 79 AHRS
Figure 26

NOTE: GRS 79 Earth Magnetic Field Updates

The GRS 79 utilizes an Earth magnetic field model which is updated once every five years. The update is expected to be available from Garmin in each of the following years: 2010, 2015, and every five years thereafter, as long as the GRS 79 remains a Garmin-supported product. The G1000 NXi system alerts the operator that the magnetic field database is out of date by issuing the message “AHRS SERVICE – AHRS Magnetic-field model needs update”. Garmin will distribute update instructions when updates are available.

(b) Troubleshooting

See also "Chart 23" on page 342502110 and "Chart 24" on page 342502111.

1) Review the airframe logbook. See if any G1000 NXi or other avionics or electrical maintenance had been performed recently that might produce or have induced the trouble being observed.
2) Check power wire connections at the circuit breakers. Inspect for loose wire terminals causing intermittent power glitches. Also, check for intermittent circuit breakers.
3) Have ground power put on the aircraft.
4) Turn on G1000 NXi and record the system software level on the MFD. This will be needed if assistance from Garmin Product Support is required.
5) After the G1000 NXi system has initialized (about one minute from power on), note any Red-X’s on the displays, ALERT messages and Red-X’s on the MFD Aux – System Status page.
6) Try to verify the issue still exists before proceeding to the physical inspection below.
7) Turn off G1000 NXi and gain access to the GRS 79.
8) Inspect the physical installation of the GRS.
    a) Is the connector tight and locking slider engaged to the locking tabs on each side of the GRS connector?
    b) Is the wire harness loose and able to move around during flight? This condition may cause the wire to pull on or vibrate the connector making intermittent connections.
    c) Is the GRS mounted tight to the rack? If any doubt exists, use a screwdriver to check the tightness of the four mounting screws.
    d) Look around the GRS for any heavy objects that may not be fastened tight to the structure that could induce GRS vibration.

e) Look for evidence of water or fluid contamination in the area around the GRS.

f) Unplug the GRS connector and check for bent pins.

g) Inspect the wire harness clamp on the rear of the connector to verify it's not too tight and damaging wires. Also check for some sort of protective wire wrap between the wires and the clamp. If the wire clamp is installed upside down, it has sharp edges that can cut into the wires.

h) Verify the locking slider spring is strong enough to keep the slider in the locked position by cycling the slider.

i) Ensure that no cell phone or other device using cell phone technology is turned on (even in a monitoring state) in the cabin.

j) Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements.

k) Check PFD1 Alert Window for PFD1/2, MFD or GRS configuration, software or failed data path error messages. Correct any errors before proceeding.

l) Ensure GRS 79 unit connecter is secure and proper wire harness strain relief is provided.

m) Ensure the GRS 79 is fastened down tightly in its mounting rack and that the mounting rack is not loose.

NOTE: Do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be releveled and and GRS 79 recalibrated.

9) Cycle GRS 79 power to restart initialization.

10) Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important for an ATTITUDE FAIL that appears during ground operation only.

11) Perform "Engine Run-Up AHRS Vibration Test" on page 342502195 to check if engine vibration is causing the GRS 79 to invalidate outputs or reset.

12) Perform "Magnetometer Interference Test" on page 342502196 to determine if there is significant magnetic interference from some device on the aircraft.

13) Replace GRS 79.

14) If problem persists, replace the GRS 79 Configuration module.

15) If the condition is not resolved by following these instructions, contact Garmin Product Support for additional assistance. In rare cases, a Garmin Field Service Engineer may need to visit your facility to retrieve the fault logs stored in the GRS to determine if the fault is in the GRS or the aircraft.

(c) Removal

1) Disconnect the AHRS connector.

2) Remove the four Phillips thumbscrews with a screwdriver and set them aside.

3) Gently lift the GRS 79 from the mounting plate.

NOTE: If the mounting plate itself is removed or loosened, the GRS 79 must be recalibrated upon reinstallation.

(d) Installation

1) Place the GRS 79 on the mounting plate, ensuring the orientation is correct.

2) Fasten the unit to the plate using the Phillips thumbscrews.

3) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 79.

(e) Return to Service

1) Original GRS 79 is Reinstalled

If the original GRS 79 was reinstalled, then no software loading is required.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

If the GRS rack was not removed or loosened, continue to "GRS/GMU Test" on page 342502198.

If the GRS rack was removed or loosened, continue to "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190.

2) Original GRS 79 Installed in Opposite Locations for Troubleshooting

If the original GRS1 and GRS2 are installed in opposite locations (GRS1 and GRS2 in opposite unit racks), no software loading is required.

If the GRS rack(s) was not removed or loosened, continue to "GRS/GMU Test" on page 342502198.

If the GRS rack(s) was removed or loosened, continue to "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190.

3) New, Repaired, or Exchanged GRS 79 is Installed

If a new, repaired, or exchanged GRS 79 unit is installed then software must be loaded and the unit must be calibrated. Continue to "G1000 NXi Software/Configuration Procedure" on page 342502152 for software loading. Then, proceed as follows:

If the GRS rack(s) was not removed or loosened, continue to "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190.

If the GRS rack(s) was removed or loosened, continue to "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190.

4) New GRS 79 Configuration Module is Installed

If the GRS 79 configuration module was replaced, no software loading is required. Continue to "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190.

CHART 23
G1000 NXI - TROUBLESHOOTING - GRS 79 (AHRS) / GMU 44(B)

Symptom	Recommended Action
AHRS does not complete initialization	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important if this issue appears during ground operation only. • Calibrate the GRS 79. • Check GRS 79 configuration module wiring for damage. • Check GRS 79 connector for bent pins. ✓ If no damage can be found, replace GRS 79 configuration module. ✓ If problem persists, replace the GRS 79.
Attitude appears unstable	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Ensure the four GRS 79 mounting screws are tight. Finger tight is not sufficient, a screwdriver must be used to verify. • Ensure mounting rack and airframe shelf are secure and all hardware and brackets are present (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Ensure GRS 79 connector is securely fastened and proper strain relief is provided. • Remove GRS 79 connector and verify there are no bent pins. • Replace the GRS 79. • Contact Garmin for further troubleshooting if required.

CHART 24 (Sheet 1 of 2)
G1000 NXi - MESSAGE ADVISORIES - GRS 79 / GMU 44(B)

Failure Message	Cause	Solution
MANIFEST – GRS1 software mismatch. Communication halted. MANIFEST – GRS2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GRS 79.	Load correct software version. See G1000 NXi Software/ Configuration Procedures, below.
AHRS1 SRVC – AHRS1 magnetic-field model needs update. AHRS2 SRVC – AHRS2 magnetic-field model needs update.	The AHRS magnetic field model in the affected unit is out-of-date. Appears on ground only.	Load updated AHRS magnetic field file. See Garmin SB 533, latest revision.
GEO LIMITS – AHRS1 too far north/south, no magnetic compass. GEO LIMITS – AHRS2 too far north/south, no magnetic compass.	No magnetic compass information available due to airplane being too far north or south.	Operate the aircraft only within the geographic limits specified in the POH.
AHRS1 TAS – AHRS1 not receiving airspeed. AHRS2 TAS – AHRS2 not receiving airspeed.	The specified GRS 79 is not receiving airspeed from the GDC 72.	Check GRS/GDC interconnect for faults. Replace the GDC 72: If problem persists, replace the GRS 79.
AHRS1 GPS – AHRS1 not receiving backup GPS information.	The GRS 79 #1 is not receiving backup GPS information from either GIA 64(W).	Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. Check GPS status for GIA 1 and 2 on MFD - AUX GPS STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting chart. Troubleshoot GIA1/2 –GRS1 wiring for faults per "Failed Path Messages." Replace the GRS 79 #1.
AHRS1 GPS – AHRS1 operating exclusively in no-GPS mode.	The GRS 79 #1 is operating in the absence of GPS.	
AHRS1 GPS – AHRS1 not receiving any GPS information.	The GRS 79 #1 is not receiving GPS data from the GPS receivers.	
AHRS1 GPS – AHRS1 using backup GPS source.	The GRS 79 #1 is using the backup GPS data path.	

CHART 24 (Sheet 2 of 2)
G1000 NXi - MESSAGE ADVISORIES - GRS 79 / GMU 44(B)

Failure Message	Cause	Solution
AHRS2 GPS – AHRS2 not receiving backup GPS information.	The GRS 79 #2 is not receiving backup GPS information from either GIA 64(W).	Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. Check GPS status for GIA 1 and 2 on MFD - AUX GPS STATUS page. If one or both GPS receivers cannot acquire a position lock, see GPS troubleshooting chart. Troubleshoot GIA1/2 –GRS2 wiring for faults per “Failed Path Messages.” Replace the GRS 79 #2.
AHRS2 GPS – AHRS2 operating exclusively in no-GPS mode.	The GRS 79 #2 is operating in the absence of GPS.	
AHRS2 GPS – AHRS2 not receiving any GPS information.	The GRS 79 #2 is not receiving GPS data from the GPS receivers.	
AHRS2 GPS – AHRS2 using backup GPS source.	The GRS 79 #2 is using the backup GPS data path.	
AHRS MAG DB – AHRS magnetic model database version mismatch.	The G1000 NXi has detected a magnetic model database version mismatch.	Load updated AHRS magnetic field file in both units. See Garmin SB 533, latest revision.
MANIFEST – GMU1 software mismatch. Communication halted. MANIFEST – GMU2 software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the specified GMU 44(B).	Load the correct software version. See G1000 NXi Software/Configuration Procedure, below.
HDG FAULT – AHRS1 magnetometer fault has occurred. HDG FAULT – AHRS2 magnetometer fault has occurred.	A fault has occurred in the specified magnetometer; heading will be flagged invalid.	Check GMU 44(B)/GRS 79 interconnect for faults. Replace GMU 44(B). If problem persists, replace affected GRS 79(B).

(13) GMU 44(B) Magnetometer

See "Figure 27". In S/N's 4636720–4636775, GMU 44 is installed. In S/N's 4636776 and up, GMU 44(B) is installed.

(a) Description

The GMU 44(B) provides horizontal and vertical magnetic field information to the GRS 79 AHRS. This allows heading to be calculated and provides assistance during AHRS alignment. The GMU is mounted under the dorsal fairing atop the aft fuselage at F.S. 282.372.

GMU 44(B) Magnetometer
Figure 27

(b) Troubleshooting

See "Chart 23" on page 342502110 and "Chart 24" on page 342502111.

(c) Removal

1) Remove the dorsal fairing to expose the magnetometers.
2) Disconnect the wiring harness(es) from the aft fuselage harness.
3) Unscrew and remove the three brass screws and washers securing the magnetometer to its mounting bracket and remove the magnetometer(s).
4) Carefully lift the GMU from the rack.

(d) Installation

CAUTION: THE MAGNETOMETERS ARE SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Lower the GMU into the rack and secure the plate with the three Phillips screws.
3) Connect wiring harness(es) to the aft fuselage harness.
4) Reinstall the dorsal fairing.

NOTE: Ensure correct hardware is used when reinstalling vertical fin dorsal fairing over magnetometers.

(e) Return to Service

1) Original GMU is Reinstalled

If the original GMU was reinstalled, then no software loading is required. Continue to "GRS/GMU Test" on page 342502198.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) Original GMU Installed in Opposite Locations for Troubleshooting

If the original GMU1 and GMU2 are installed in opposite locations (GMU1 and GMU2 in opposite unit racks), no software loading is required. However, performing "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190. is required, then, continue to "GRS/GMU Test" on page 342502198.

3) New, Repaired, or Exchanged GMU is Installed

If a new, repaired, or exchanged GMU unit is installed then software must be loaded. Continue to "G1000 NXi Software/Configuration Procedure" on page 342502152, then continue to "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190.

(14) GDL 69eA / 69A SXM Satellite Datalink Unit

See "Figure 28". In S/N's 4636720–4636775, GDL 69eA is installed. In S/N's 4636776 and up, GDL 69A SXM is installed.

(a) Description

The GDL 69eA / 69A SXM provides XM Radio weather and music entertainment through means of a dedicated satellite datalink. The GDL is mounted behind the PFD1. Power to the GDL is received from the #1 avionics bus. The GDL sends weather data through the HSDB bus to the MFD, where the datalink interface is controlled. Digital audio is sent directly to the GMA 347 or GMA 350c Audio Panel.

(b) Troubleshooting

See "Chart 25" on page 342502115 and "Chart 26" on page 342502116.

GDL 69eA / 69A SXM Datalink Unit
Figure 28

(c) Removal

1) Remove PFD1 per "Removal" on page 34250268.
2) Unlock the GDL handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GDL.
4) Gently remove the GDL from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GDL into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GDL body using the Phillips screw.
5) Reinstall the PFD1 per "Installation" on page 34250268.

(e) Return to Service

1) Original GDL is Reinstalled

No software or configuration loading is required if the removed GDL is reinstalled. Continue to "GDL 69eA / 69A SXM Test" on page 342502199.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GDL is Installed

If a new, repaired, or exchanged GDL is installed, the correct software and configuration files must be loaded to the unit, then the XM Satellite Radio subscription must be reactivated, see "System Upload" on page 342502158, then continue to "GDL 69eA / 69A SXM Test" on page 342502199.

CHART 25
G1000 NXi - TROUBLESHOOTING - GDL 69eA / 69A SXM

Symptom	Recommended Action
No XM audio is heard No XM weather information is displayed.	Ensure the following items are not preventing the audio panel from distributing XM audio (see also applicable Meridian G1000 NXi Pilot's Guide). Verify XM volume is not muted on the AUX–XM RADIO page on the MFD. Ensure nothing is plugged into the Stereo Input Jack. Verify the COM squelch is not open. Verify the ICS squelch is not open. Verify the marker beacon tones are not being received. Verify the headphone (if equipped) volume is turned up. Go to the AUX – SYSTEM STATUS page on the MFD and ensure unit is online. If a red X is present, verify the unit is receiving power at the rack connector. Ensure there are no GDL alerts in the alert window. If there is an Alert for software or configuration error or mismatch, reload the file noted in the Alert. Restart the PFDs and MFD in configuration mode and go to the GDL page. Verify unit is active. Verify the Signal number is "2" or "3". If it is "0" or "1" check the GDL antenna and cabling for faults. Reseat the GDL to verify the coax connector is fully seated. If unit is not active, contact XM Customer service at 1-800-985-9200 to have a refresh signal sent to your unit.

NOTE: You will need to provide them the Audio Radio ID (and Data Radio ID number for XM weather) numbers. Also verify with XM that the correct Weather package (Aviator Lite or Aviator) is on the account, and that no traffic service has been activated against that Radio ID. The unit must be on for approximately one hour after the request for the refresh has been sent to receive the signal.

To obtain radio IDs, power-up PFD1 in normal mode and then select XM INFO page in AUX page group. XM weather services has an 8 digit "data radio ID" and XM radio services has an 8 digit "audio radio ID" displayed on this page.

	Alternatively, you may also go to XM's website at https://care.siriusxm.com/retailrefresh_view.action#/refreshradio and enter the radio ID's to have a refresh signal sent. If there is still problems receiving weather products after performing the above step, call XM and have the account deactivated, and a new account activated to clear out any corrupt account information. Verify there is a good ground connection through the aircraft between the MFD and the GDL unit. See 51-80-00 for bonding checks. If problem persists, replace the GDL.

CHART 26
G1000 NXi - MESSAGE ADVISORIES - GDL 69eA / 69A SXM

Failure Message	Cause	Solutions
GDL FAIL – GDL has failed.	The G1000 NXi has detected a failure in the GDL.	• Replace GDL. • Check GDL antenna and cabling. • Check the GDL and MFD interconnect.
GDL CONFIG – GDL configuration error. Config service req'd.	The G1000 has detected a GDL configuration mismatch.	• Reload configuration file. See Software Load / Configuration Procedure. • If problem persists, replace master configuration module, check config module harness for faults and replace if necessary. NOTE New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The change G1000 NXi System ID number will to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.
MANIFEST – GDL software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GDL.	• Load correct software version. See Software Load / Configuration Procedure.

(15) FS 510 Bluetooth® and Wi-Fi Connectivity (Optional)

See "Figure 29".

(a) Description

The FlightStream 510 is an SD card-sized unit which is inserted into the lower SD card slot on the face of the MFD. It provides a secure (password protected) Bluetooth® and Wi-Fi interface to the G1000 NXi to provide wireless database updating via the Garmin Pilot app, situational data available to mobile apps, GPS and attitude information, ADS-B (Traffic and FIS-B), Flight Plans, PVT, AHRS, TIS-B, FIS-B, Data logging, and SXM Weather.

(b) Troubleshooting

See "Chart 27" on page 342502118.

(c) Removal

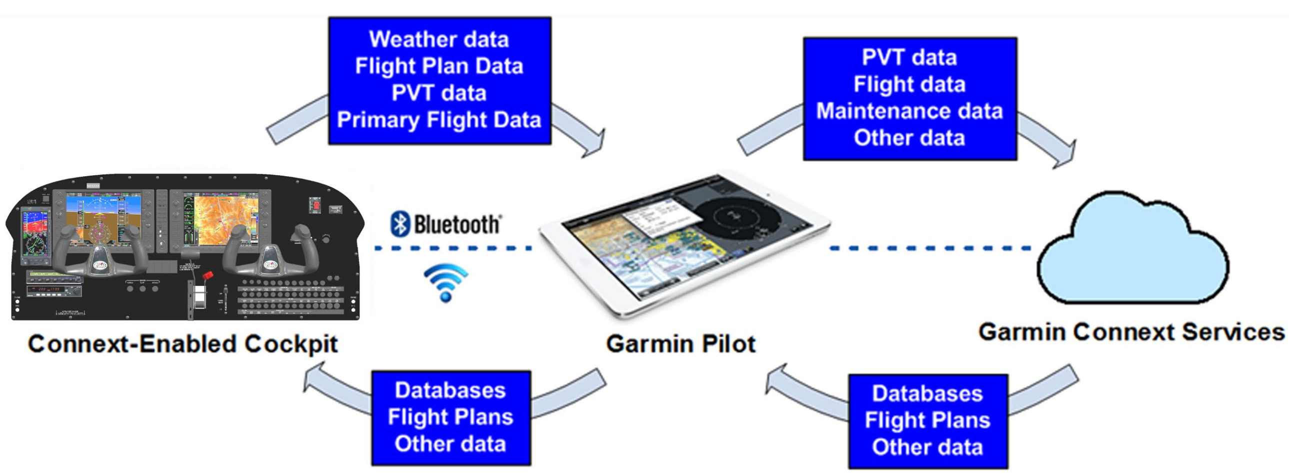
1) Press and release unit.

2) Remove unit from card slot.

(d) Installation

1) Place unit in card slot.

2) Press unit into card slot until you feel unit engage slot.



FS 510 (Optional)
Figure 29

CHART 27
TROUBLESHOOTING - FS 510

Problem	Action
Flight Stream Not Found	Check Flight Stream wiring and power. Check that Flight Stream is not obstructed by RF blocking material. Check that the control/ display is in RF range of the Flight Stream. Check that pairing mode is enabled.
Flight Stream connection lost.	Check that the control/ display is in RF range of the Flight Stream. Check that Flight Stream is not obstructed by RF blocking material. Remove the Bluetooth® pairing from the portable electronic device and the Flight Stream, and then attempt to pair the device again.

(16) GSR 56 Iridium Satellite Transceiver

Optional

See "Figure 30".

(a) Description

The optional GSR 56 Iridium Satellite Transceiver provides airborne Iridium Satellite Telephone and SMS Messaging service. Iridium telephone and text meassaging are available to the flight crew through the MFD, audio panel, and headset. The Garmin GSR 56 is connected to the #2 GIA 64 and GMA 350c. Power is received from the Avionics Bus #1.

GSR 56 Iridium Transceiver
Figure 30

(b) Troubleshooting

See "Chart 28" on page 342502120.

(c) Removal

1) Remove the cabin rear closeout panel to gain access to the upper aft avionics shelf.
2) Loosen the GSR 56 unit by turning the ratchet mechanism located at the front of the unit counter-clockwise until it drops free of the locking pawl on the unit.
3) Gently remove the GSR 56 from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GSR 56 into the rack and engage the connectors.
3) Lift the ratchet mechanism to allow the collar to engage the locking pawl on the unit.
4) Gently turn ratchet mechanism clockwise to secure the GSR 56 unit into place.
5) Reinstall the cabin rear closeout panel.

(e) Return to Service

The GSR 56 does not require software or configuration loading if replaced. Instead, Garmin and/or Iridium need to be contacted to establish service.

See "Chart 28" on page 342502120.

CHART 28
TROUBLESHOOTING / MESSAGE ADVISORIES - GSR 56

Problem	Action
No communication with GSR 56.	Check power wiring and pin out. Verify correct communication port setting on display/control device.
No or low-quality signal.	Ensure the Iridium antenna has an unobstructed view of satellite constellation. Check the antenna cable and connectors. Verify antenna ground plane is adequate.
No audio output.	Check wiring from GSR 56 to audio panel or GDL. Verify subscription with Garmin Iridium Services, http://fly.garmin.com or 1-866-739-5687.
Unable to make a phone call.	Verify subscription with Garmin Iridium Services, see above. Verify signal quality is adequate. Verify communication between GSR 56 and the display/control device.

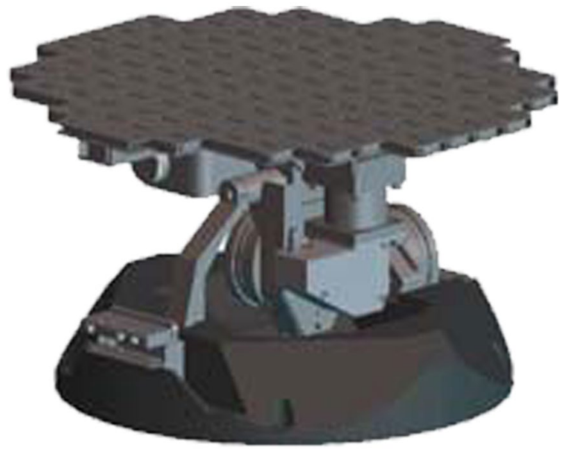
Failure Message	Cause	Solution
GSR1 FAIL – GSR1 is inoperative.	The G1000 NXi has detected a failure in the GSR 56.	Check wiring for faults per “Failed Path Messages.” Replace the GSR 56.

(17) GWX 68 / 75 Weather Radar

See "Figure 31". In S/N's 4636720–4636775, GWX 68 is an optional installation. In S/N's 4636776 and up, GWX 75 is an optional installation.

(a) Description

The GWX 68 / 75 Airborne Weather Radar provides weather radar data output to the GDU 1250A MFD. The GWX is mounted in the right wing radar pod. Power to the GWX is received from the avionics No. 1 bus. Data received from the GWX is routed through the GDL 69eA / 69A SXM data link unit to the MFD via high-speed data bus (Ethernet). If the optional GDL is not installed, the GWX Ethernet is connected directly to the MFD.



GWX 68 / 75 Weather Radar
Figure 31

(b) Troubleshooting

See "Chart 29" on page 342502122 and "GWX 68 / 75 Test" on page 342502200.

(c) Removal

1) Remove screws and washers (13 ea.) and remove radome.

2) Disconnect backshell assembly from unit.

3) Use a 3/16” hex drive tool to remove each of the four mounting screws.

(d) Installation

1) Inspect connector for damaged pins.

2) Hold unit flush with the radar mount.

3) Use a 3/16” hex drive tool to tighten each of the four mounting screws. Torque for 1/4-28 fasteners per Gap Conditions Between Parts Attached with Threaded Fasteners under Torque Requirements in 91-10-00, except final torque not to exceed 60 in.lbs.

4) Connect backshell assembly to unit.

5) Install radome and secure with screws and washers (13 ea.).

(e) Return to Service

1) Original GWX Reinstalled

No software or configuration loading is required if the removed GWX is reinstalled, continue to "GWX 68 / 75 Test" on page 342502200.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GWX Installed

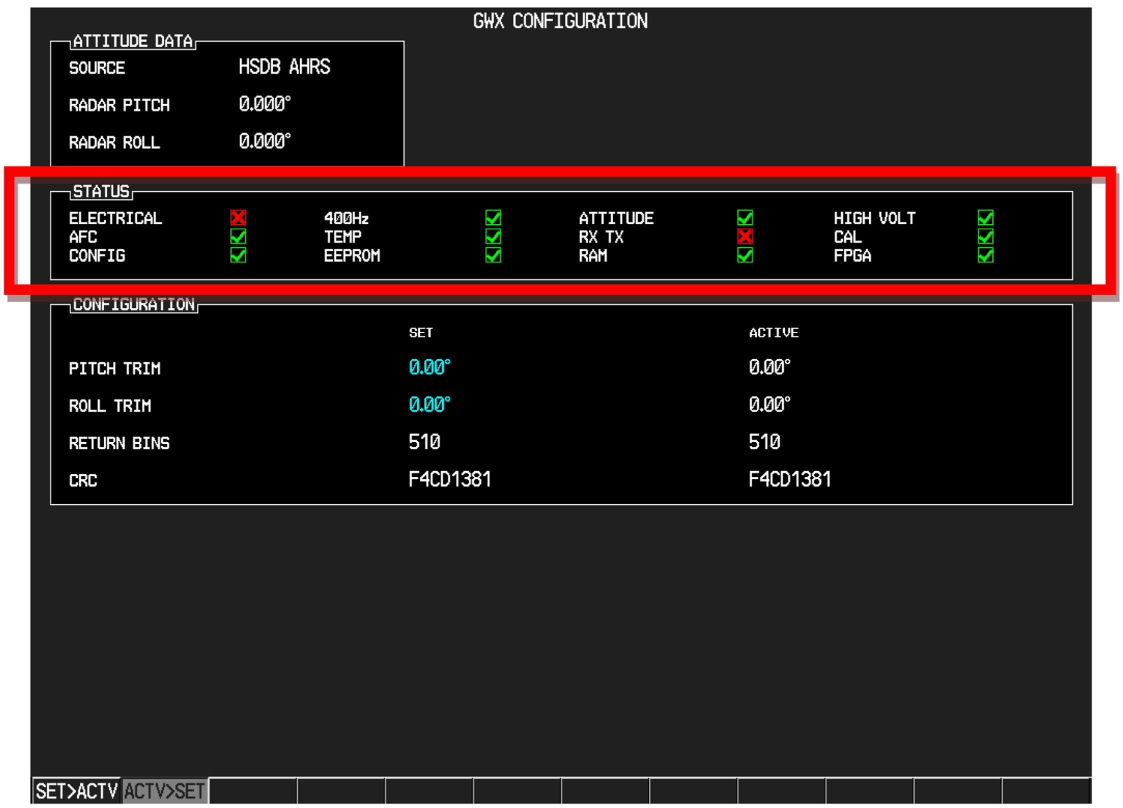
If a new, repaired, or exchanged GWX is installed, the correct software and configuration files must be loaded to the unit.

See "G1000 NXi Software/Configuration Procedure" on page 342502152 and then continue to "GWX 68 / 75 Test" on page 342502200.

CHART 29 (Sheet 1 of 5)
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68 / 75

WARNING: BEFORE ENERGIZING THE EQUIPMENT, BE SURE MICROWAVE RADIATION SAFETY PRECAUTIONS INCLUDING BOTH FUEL AND PERSONNEL SAFETY CONSIDERATIONS HAVE BEEN OBSERVED. THESE INCLUDE CLEARING ALL PERSONNEL TO AN AREA BEYOND THE MAXIMUM PERMISSIBLE EXPOSURE LEVEL (MPEL) BOUNDARY. THE MPEL FOR THE GWX IS 11 FEET AND FOR THE GWX 75 IS 12 FEET.

GWX Configuration Page Data Fault Indications



NOTE: The fault indications are reset if PFD1 is transitioned from normal mode to configuration mode. It will return if the fault is persistent. It may not return if the fault is intermittent. The fault indications also reset with a power cycle to the unit.

NOTE: Transitioning the MFD to configuration mode will not reset the fault indications.

CHART 26 (Sheet 2 of 5)

G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68 / 75

STATUS INDICATOR BOX	FAULT INDICATION	FAULT INDICATION DESCRIPTION	CORRECTIVE ACTION
Electrical		Voltages within the unit are out of range.	• Cycle unit twice to confirm problem. • Replace the GWX if condition remains.
400 Hz			• N/A for G1000 NXi installation: this box should always be green.
High Volt		Monitoring for voltage exceeding 33V at the power input to the GWX. Transitions unit to standby mode if voltage exceeds 33V.	• Check voltage at input unit (should be less than 33V). If over 33V, troubleshoot aircraft power system. • Check if condition occurred due to aircraft system power spike due to other non-G1000 NXi electrical equipment loads. • Correct issues that cause aircraft electrical system power spikes. • If problem persists, replace the GWX. • Cycle power on unit only to see if fault remains, if it does, replace the GWX.
AFC		Automatic frequency control within the unit has a fault, will make unit enter standby.	• If problem persists, replace the GWX.
TEMP		Unit temp exceeds 85°C.	• If condition occured during flight, replace the GWX. • If condition occurs on the ground due to excessive heat inside radome, turn unit off and allow it to cool before further use. • If the condition occurs on the ground and the temperature is normal in the radome, replace the GWX.
RX TX		Magnetron fault within the unit, will make unit enter Standby mode.	• Ensure the GWX software version 2.10 or later is installed. • If problem peresists, replace the GWX.

CHART 26 (Sheet 3 of 5)
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68 / 75

STATUS INDICATOR BOX	FAULT INDICATION		FAULT INDICATION DESCRIPTION	CORRECTIVE ACTION
CAL			Bad factory calibration data.	• Replace the GWX.
CONFIG			GWX and GDU configuration do not match.	• Troubleshoot per GWX config message below.
EEPROM			Internal EEPROM component failed.	• Replace the GWX.
RAM			Internal RAM component failed.	• Replace the GWX.
FPGA			Internal FPGA component failed.	• Replace the GWX.
ATTITUDE			The unit does not have a valid attitude source via A429 databus.	• Check the AHRS source for proper operation. • If problem persists, replace the GWX.

CHART 26 (Sheet 4 of 5)
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68 / 75

Symptom	Possible Cause	Resolution
Radar audibly noisy during ground operation.	Excessive gear train noise or radome interference.	Remove radome. Inspect interior surfaces of the radome for contact with the radar dish. Correct interference by adjusting or replacing radome. If no radome interference is found, perform the following: Turn on the G1000 NXi. Select the radar page on the MFD. Press the following softkeys in the following order: 7, 9, 9, 7. Press the MODE softkey Press the TEST softkey. The radar dish will begin scanning, but it will not be transmitting. Test the horizontal and vertical modes to exercise the gear train and evaluate for noise. Replace the GWX if the gears are excessively noisy.
Radar fails to make a full vertical or horizontal sweep on the screen.	Radome Interference or GWX failure.	Remove the radome and inspect interior surfaces for contact with the radar dish. Correct interference by adjusting or replacing the radome. If problem persists, replace the GWX.

CHART 26 (Sheet 5 of 5)
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - GWX 68 / 75

Failure Message	Cause	Solution
GWX CONFIG – GWX configuration error. Configuration service required.	The G1000 NXi has detected a GWX configuration mismatch.	Load GWX configuration files. Replace GWX. If problem persists, replace master configuration module(1), check config module wiring for faults and replace if necessary.
GWX FAIL – GWX is inoperative.	The G1000 has detected a failure in the GWX.	Check Ethernet connection between the GWX and GDL69eA / 69A SXM for faults. Replace the GWX.
GWX SERVICE – Needs service. Return unit for repair.	The G1000 NXi has detected a failure in GWX.	Replace the GWX.
MANIFEST – GWX software mismatch. Communication halted.	The system has detected an incorrect software version loaded in the GWX.	Load correct software version. See G1000 NXi Software/ Configuration Procedure, below.

(1) New Terrain/Obstacle cards, Jeppesen Aviation Database and other optional features (i.e. TAWS unlock card) will need to be replaced if the master configuration module is changed. The G1000 NXi System ID number will change to a new number when installing a new master config module. The old Terrain and other cards will no longer work as they will remain locked to the old System ID number.

(18) GTS 825 Traffic Advisory System

Optional

See "Figure 32".

(a) Description

The optional GTS 825 Traffic Advisory System provides real-time traffic information to the PFD (and, indirectly, to the MFD). The Garmin GTS 825 connects with the #1 GDU 1050 and communicates with the MFD through an HSDB connection. Power is received from the Avionics Bus #1.

GTS 825 Processor
Figure 32

(b) Troubleshooting

See "Chart 30" on page 342502128.

(c) Removal

1) Remove the cabin rear closeout panel to gain access to the lower aft avionics shelf. See 25-20-00.
2) Disconnect the GTS 825 connectors.
3) Loosen the GTS 825 unit by unscrewing the knurled retainers.
4) Move the retainer arms downward. Gently remove the GTS 825 from its rack.

(d) Installation

1) Insert the GTS 825 into the rack and lift each retainer arm to engage lip.
2) Gently screw retainers to secure the GTS 825 unit into place.
3) Visually inspect the connectors and pins for damage. Repair any damage. Connect the connectors to the unit.
4) Reinstall the cabin rear closeout panel. (See 25-20-00.)

(e) Return to Service

1) Original GTS 825 Reinstalled

No software or configuration loading is required if the removed GWX 68 / 75 is reinstalled. Continue to "GTS 825" on page 342502207 under LRU Test Procedures.

NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

2) New, Repaired, or Exchanged GTS 825 Installed

If a new, repaired, or exchanged GTS 825 is installed, the correct software and configuration files must be loaded to the unit. See "G1000 NXi Software/Configuration Procedure" on page 342502152 and then continue to "GTS 825" on page 342502207 under LRU Test Procedures.

CHART 30 (Sheet 1 of 2)
ADVISORY MESSAGES - GTS 825

Advisory Message	Possible Cause	Recommended Actions
CALIBRATION DATA FAULT	Stored factory calibration parameters are invalid.	Replace the unit.
CONFIGURATION DATA FAULT	Stored system configuration parameters are invalid or Mode S address is invalid (All 0's or F's). Fault will persist until configuration is corrected.	Reconfiguration the GTS 825. If problem persists, replace the unit.
FPGA FAULT	Check of the FPGA image failed. Fault will persist until valid FPGA image is loaded.	If upload of FPGA image was recently attempted, retry the upload. Otherwise, replace the unit.
ROM FAULT	Internal non-volatile memory failure, or invalid data image detected.	If upload of audio image or IGRF magnetic field image was recently attempted, retry the upload. Otherwise replace the unit.
EXECUTION FAULT	CPU execution fault has occurred.	Cycle power and retry self test If fault persists, replace the unit.
ELECTRICAL FAULT	One of the internal electrical voltages are out of range. Fault will persist until power is cycled.	Check aircraft power supply. If fault persists, replace the unit.
WHISPER SHOUT FAULT	Transmitted power is out of tolerance.	Check cable loss configuration, antenna installation and all cable connections and retry self test. If fault persists, replace the unit.
TRANSMIT POWER FAULT	One of the internal transmitter power source voltages are out of range. Fault will persist until power is cycled.	Check aircraft power supply. If fault persists, replace unit.
1030 MHZ FREQUENCY SOURCE FAULT	Transmit Frequency synthesizer is not locked.	Cycle power and retry self test. If fault persists, replace unit.
1090 MHZ FREQUENCY SOURCE FAULT	Receive Frequency synthesizer is not locked.	Cycle power and retry self test. If fault persists, replace unit.
RECEIVER CALIBRATION FAULT	N/A	Check antenna installation and all cable connections and retry self test. Ensure that self test occurs in area free of buildings and large objects that can reflect signals. If fault persists, replace unit.

CHART 29 (Sheet 2 of 2)
ADVISORY MESSAGES - GTS 825

Advisory Message	Possible Cause	Recommended Actions
TRANSMITTER CALIBRATION FAULT	N/A	Check antenna installation and all cable connections and retry self test. Ensure that self test occurs in area free of buildings and large objects that can reflect signals. If fault persists, replace unit.
BAROMETRIC ALTITUDE INPUT FAULT	Own ship barometric altitude calculation is invalid or has timed out.	Check wiring to source of barometric altitude and ensure that source is operating. Fault will clear as soon as valid barometric altitude data is received.
MAIN BOARD TEMPERATURE FAULT	Main board temperature or RF receiver temperature is greater than 90° Celsius or less than -60° Celsius.	Fault will persist until internal temperature returns to acceptable range.
TCAS EQUIPAGE TIMEOUT FAULT	TCAS Equipage data is not being received or has timed out for 800ms.	Check wiring to TCAS Equipage data source and ensure that source is operating. Fault will clear as soon as valid TCAS Equipage data is received.

(19) Configuration Module

See "Figure 33".

The configuration module is located in the backshell of the connector it is terminated to.

(a) Removal

1) Disconnect connector from LRU.

2) Remove 2 screws (8) from cover (7) and remove cover.

3) Unplug connector from configuration module (1).

4) Remove configuration module.

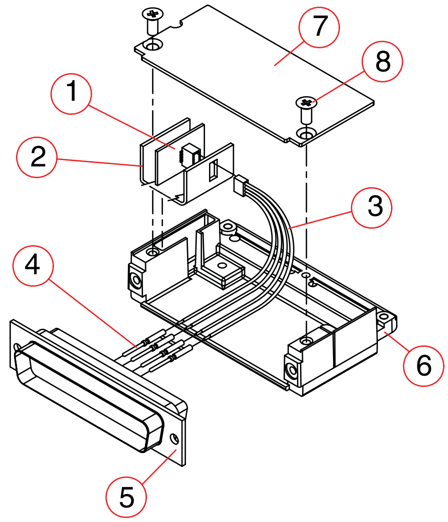
(b) Installation

1) Inspect connector for damaged pins (4).

2) Place configuration module (1) in position.

3) Insert connector into configuration module (1).

4) Assembly of the connector is the reverse of disassembly.



Configuration Module
Figure 33

(c) Return to Service

1) If a GRS 79 AHRS Configuration Module is replaced:

All three GRS 79 / GMU 44(B) calibration procedures must be performed. Proceed to "GRS 79 / GMU 44(B) Calibration Procedures" on page 342502190.

2) If a GDC 72 Configuration Module is replaced:

Configuration settings must be reloaded to the GDC 72. Continue to "G1000 NXi Software/Configuration Procedure" on page 342502152, then perform "GDC 72 Air Data Computer Tests" on page 342502188.

3) If the GEA 71B Configuration Module is replaced:

Perform "GEA 71B Test" on page 342502186.

4) If the Master Configuration Module is replaced:

a) Start the G1000 NXi system in configuration mode.

b) Go to the Configuration Upload Page on the PFD.

c) Press the UPDT CFG softkey.

5) If both the PFD and Master Configuration Module is replaced:

The G1000 NXi system (except GRS 79/GMU 44(B) and GDC 72) must be reconfigured, continue to "G1000 NXi Software/Configuration Procedure" on page 342502152.

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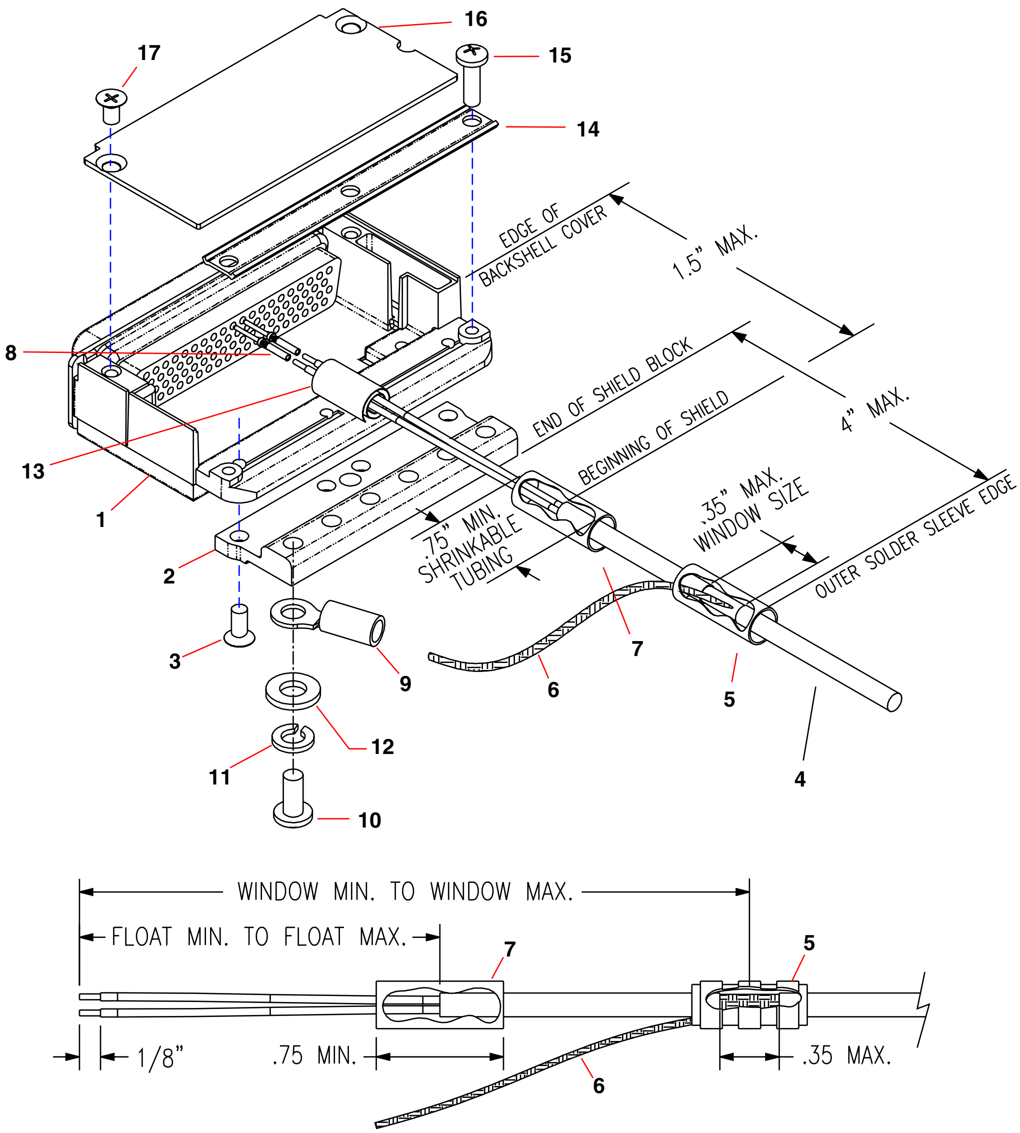
(20) Shield Block Installation

See "Figure 34" on page 342502133.

Most G1000 NXi connectors employ a shield block grounding system to provide necessary ground reference to shielding and/or transducers.



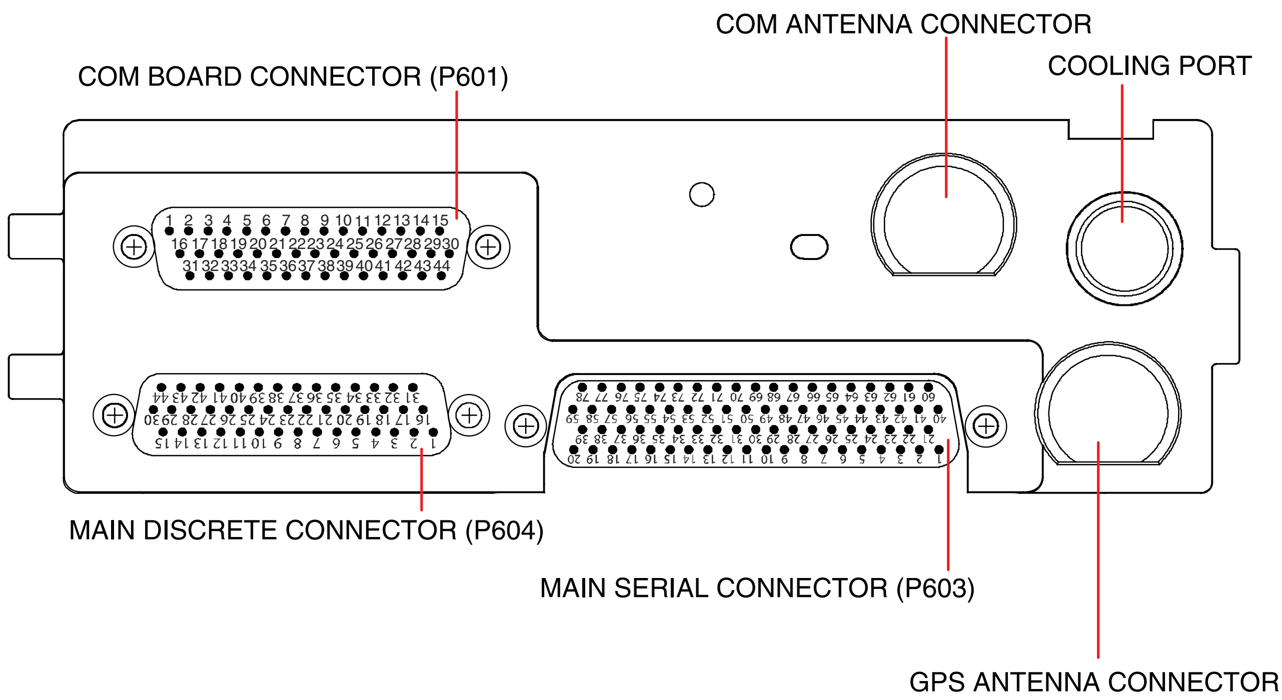
1. CAST HOUSING
2. SHIELD BLOCK(S)
3. SCREW
4. MULTIPLE CONDUCTOR SHIELDED CABLE (2–CONDUCTOR DEMONSTRATED HERE)
5. DRAIN WIRE SHIELD TERMINATION (METHOD OPTIONAL)
6. BRAID, FLAT
7. FLOATING SHIELD TERMINATION (METHOD OPTIONAL)
8. PINS
9. RING TERMINAL, #8, INSULATED
10. SCREW
11. SPLIT WASHER
12. FLAT WASHER
13. SILICON FUSION TAPE
14. STRAIN RELIEF
15. SCREW
16. LID
17. SCREW

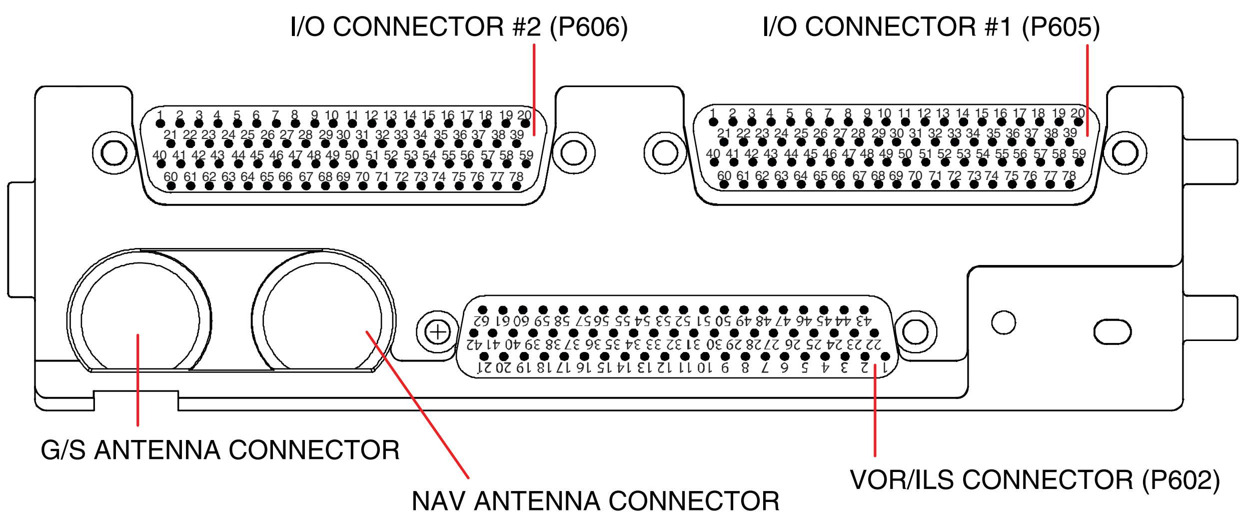
Shield Block Installation to Backshell
Figure 34

(21) Backshell Connectors

See "Figure 35" on page 342502135.

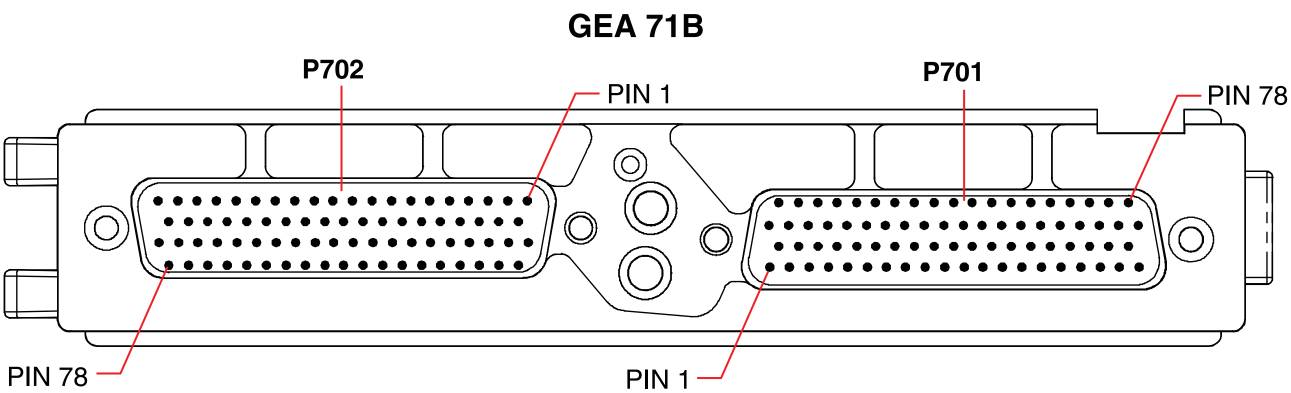
GIA 64(W)

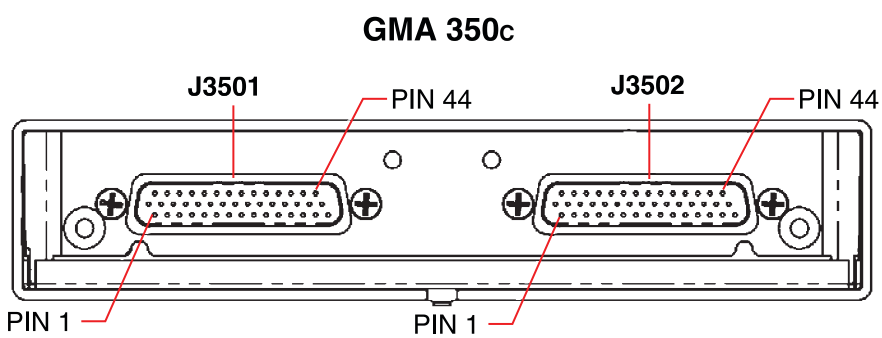




Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 35 (Sheet 1 of 5)

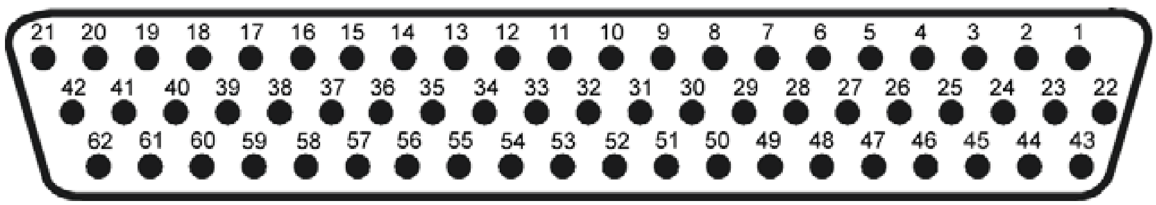




Backshell connectors as viewed with the LRU removed.

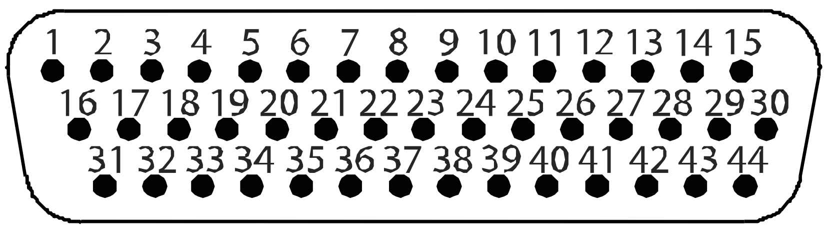
Backshell Connectors
Figure 35 (Sheet 2 of 5)

GDU 1050 / GDU 1250A



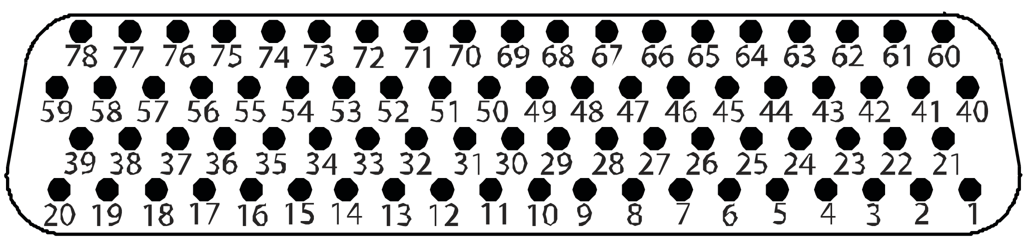
#P10001

GRS 79



#P771

GDC 72

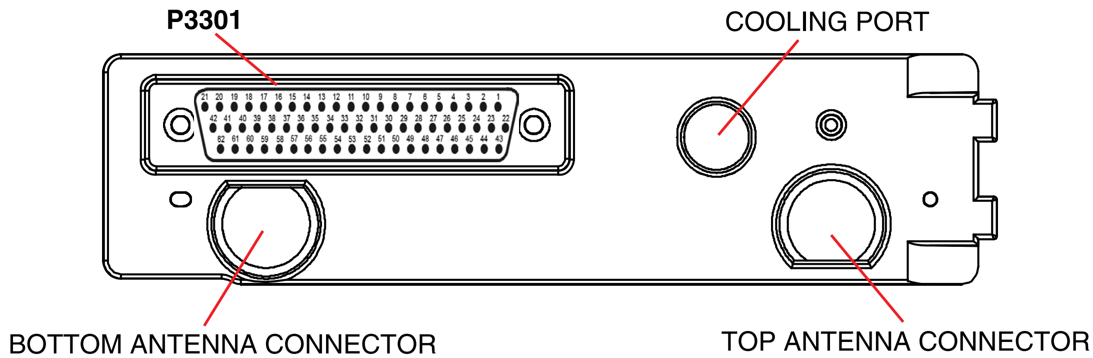


#P741

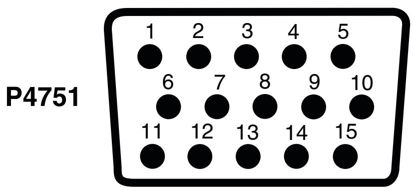
Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 35 (Sheet 3 of 5)

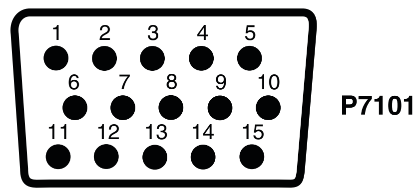
GTX 335R



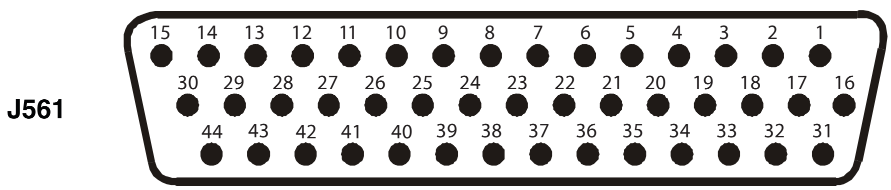
GCU 47X



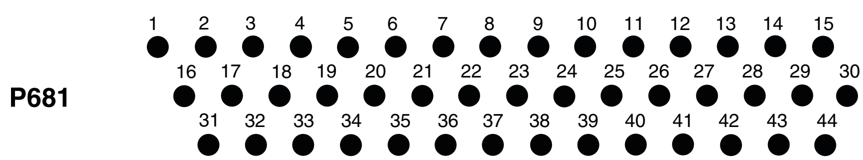
GMC 710



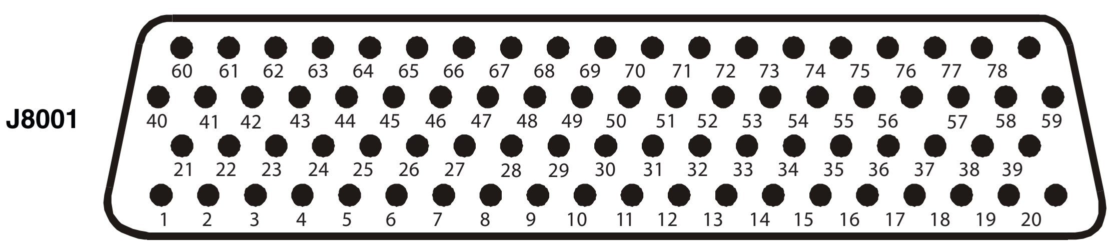
GSR 56



GWX 68 / 75



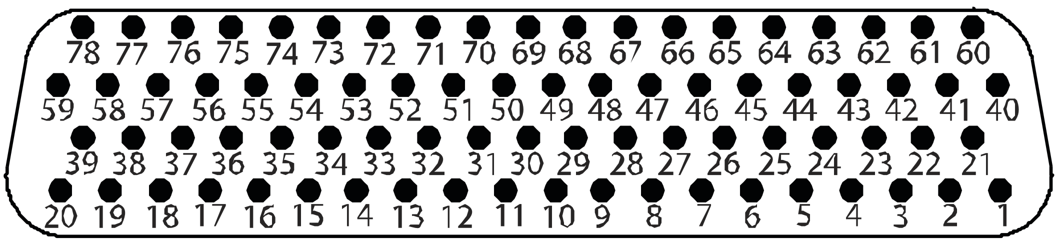
GTS 825



Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 35 (Sheet 4 of 5)

GDL 69eA / 69A SXM



P691

Backshell connectors as viewed with the LRU removed.

Backshell Connectors
Figure 35 (Sheet 5 of 5)

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H. Configuration Mode Overview

CAUTION: THE CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

The Configuration Mode exists to provide the avionics technician with a means of configuring, checking, and calibrating various G1000 NXi sub-systems. Troubleshooting and diagnostics information can also be viewed in this mode.

(1) System Configuration and Status Verification

(a) Enter the configuration mode on the PFDs and the MFD by applying power while holding ENT on each display.

(b) Verify on the "SYSTEM STATUS PAGE" that the following are valid. (Green annunciations indicate that the system is healthy): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), GWX, and GTX 2 (optional).

(c) Rotate the small FMS knob to navigate to the "SYSTEM CONFIGURATION PAGE" and verify that the following are valid. (Indicated by green annunciations): MFD1, PFD1, PFD2, GIA1, GIA2, GDL 69 (optional), GTS (optional), GWX, GTX 2 (optional), GMA 1, and GMA 1 Aux.

NOTE: For a complete description and breakdown of each Configuration Mode page, refer to the G1000 NXi Line Maintenance & Configuration Manual, Garmin P/N 190-00303-04.

(2) SET>ACTV Configuration

CAUTION: THE ACTV>SET SOFTKEY MUST BE USED WITH CAUTION! IF AN IMPROPERLY CONFIGURED UNIT IS INSTALLED, THIS SOFTKEY CAUSES THE WRONG CONFIGURATION TO REPLACE THE CORRECT AIRCRAFT CONFIGURATION.

Throughout the configuration mode pages, there are SET and ACTIVE columns for input/output settings and other parameters.

“SET”: Refers to a setting or group of settings that reside in PFD Internal Memory and/or the Master Configuration Module.

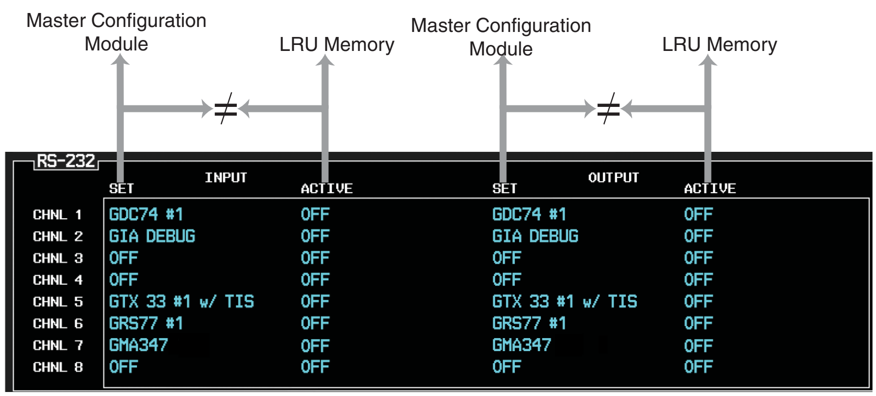
ACTIVE: Refers to an ‘active’ setting or parameter currently being used by the LRU. LRUs store the ‘active’ settings within internal memory.

Data can be manually copied from one column to the other (and consequently from PFD memory to the LRU memory and vice-versa) by using the following two softkeys, when available:

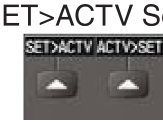
- **SET>ACTV** (read ‘Set to Active’) softkey: Allows the installer to send the information in the SET column (data stored in the master config module) to the ACTV column (data used by LRU).
- **ACTV>SET** (read ‘Active to Set’) softkey: Causes the LRUs current settings to be copied to the master configuration module as SET items.

In the first example shown in "Figure 36", the SET columns do not match the ACTIVE columns. The inequality between SET and ACTIVE indicates a configuration mismatch. By pressing the SET>ACTV softkey, this copies the SET column to the LRU unit's configuration memory. The settings then become the ACTIVE settings for the LRU being configured.

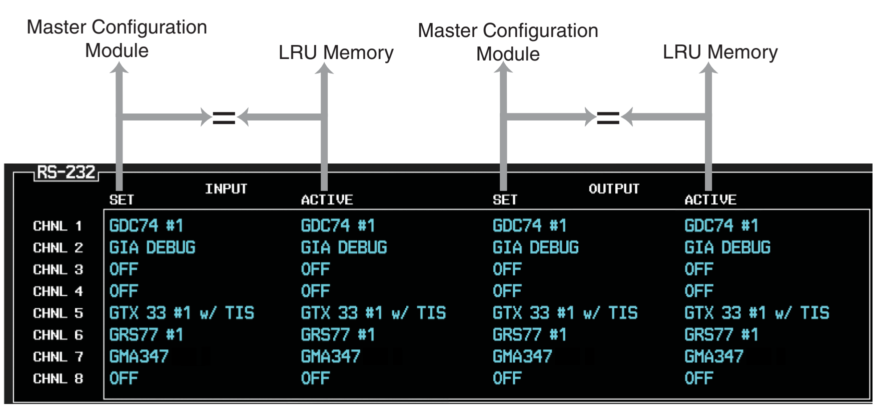
Configuration Mismatch



SET>ACTV Softkey



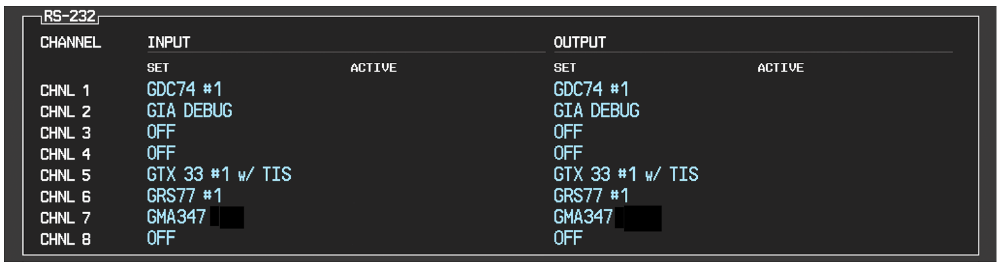
Configuration Correct



SET>ACTV Configuration
Figure 36

When troubleshooting the system, technicians can look for inequalities between SET and ACTIVE columns. Certain problems can be resolved simply by pressing the SET>ACTV softkey, which reloads settings to the specific LRU from the PFD (Note that this can also be accomplished by reloading the configuration files for the LRU, using the software loader card.).

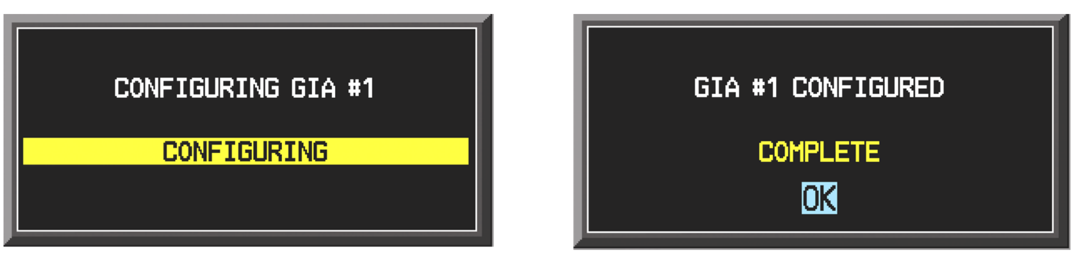
A blank active column, as shown in "Figure 37", represents loss of communication between the display and the particular unit. See troubleshooting for more details.



Blank Active Columns
Figure 37

(3) Configuration Prompts

When configuration settings are changed, the technician receives on-screen prompts and/or confirmations such as those shown in "Figure 38". Other prompts may be encountered during the configuration process.



Configuration Prompts
Figure 38

(4) Data Transmission Indicators

See "Figure 39".

Several configuration screens utilize an indicator light system to show discrete (ON/OFF) data and/or hardware component status. Unless otherwise noted, the following applies to all such status indicators:

(a) Green Light: Expected data is successfully received and is ON. A green light could also indicate that the parameter/component is working correctly.

(b) Red Light: Expected data is not received. A red light could also indicate that a parameter/ component is invalid.

(c) No Light (Black): Expected data is successfully received and is OFF, or no data is expected. A black light could also indicate that the parameter/component is not responding.

(5) Configuration Mode Navigation

Using the FMS knob, a user can navigate through different pages and page groups in the Configuration Mode. See "Chart 31" on page 342502145 For complete description and breakdown of each page, refer to the G1000 NXi Line Maintenance & Configuration manual.

Data Transmission Indicators
Figure 39

CHART 31
CONFIGURATION MODE NAVIGATION

System Page Group		
1. System Status	6. File Manager	11. System Setup
2. Date/Time Setup	7. Diagnostics Terminal	12. Manifest Configuration
3. Main Lighting	8. OEM Diagnostics	13. Maintenance Log
4. Audio Alert Configuration	9. System Configuration	
5. System Upload	10. System Data Path Config	
GDU Page Group		
1. RS-232 / ARINC 429 Config	4. Diagnostics	7. Airframe Configuration
2. GDU Status	5. Serial/Ethernet I/O	8. TAWS Configuration
3. GDU Test	6. Alert Configuration	
GIA Page Group		
1. RS-232 / ARINC 429 Config	3. GIA I/O Configuration	5. GIA Status
2. GIA RS-485 Config	4. COM Setup	6. GIA CAN Config
GEA Page Group		
1. Engine Data	2. GEA Status	3. GEA Configuration
GTX Page Group		
1. RS-232 / ARINC 429 Config	2. Transponder Configuration	
GRS Page Group		
1. AHRS / Air Data Input	2. GRS / GMU Calibration	
ADC Page Group		
1. ADC Configuration	2. GDC Configuration	
GFC Page Group		
1. GFC Configuration	2. GFC Status	
GMA Page Group		
1. GMA Configuration		
GDL Page Group		
1. GDL 69 Config		
RMT Page Group		
1. Remote Controller Status		
GWX Page Group		
1. GWX Configuration		
OTHER Page Group (Group not present unless Stormscope configured ON.)		
1. Stormscope		
CAL Page Group		
1. Fuel Temperature Calibration	2. Flaps and Trim Calibration	3. HSCM Calibration

I. G1000 NXi / GFC 700 Software Information

NOTE: The following sections provide a detailed description of loading all G1000 NXi software and configuration files, which may be excessive for individual LRU removal and replacement. If removing and replacing individual LRUs, refer to Components, above, for the necessary steps.

(1) G1000 NXi Software Image

CAUTION: IT IS CRITICAL THAT THE TECHNICIAN INSTALLS THE CORRECT SOFTWARE IMAGE PART NUMBER WHEN SERVICING THE G1000 NXI SYSTEM. APPROVED SOFTWARE DEFINITIONS FOR EACH CONFIGURATION ARE DEFINED IN THE REQUIRED EQUIPMENT LIST (REL), "CHART 41" ON PAGE 342502227.

All software and configuration files were certified by Garmin as part of the FAA-approved Type Design data. Approved software definitions for each configuration are defined in the appropriate Required Equipment List (REL), "Chart 41" on page 342502227.

G1000 NXi software and configuration files are controlled via the approved software image part number listed in the REL. This software image is loaded into the G1000 NXi using a software loader card. The installer shall create this software loader card by downloading the approved software image in accordance with the procedure provided immediately following.

CAUTION: BE CAUTIOUS WHEN USING SOFTWARE LOADER CARDS DURING MAINTENANCE. THE G1000 NXI SYSTEM IMMEDIATELY INITIALIZES THE CARD UPON POWER-UP. ON-SCREEN PROMPTS MUST BE GIVEN CAREFUL ATTENTION IN ORDER TO AVOID POTENTIAL LOSS OF DATA. ALWAYS READ THROUGH PROCEDURES GIVEN IN THIS SECTION, BEFORE ATTEMPTING TO USE THE SOFTWARE LOADER CARDS.

(2) Software Loader Card Creation

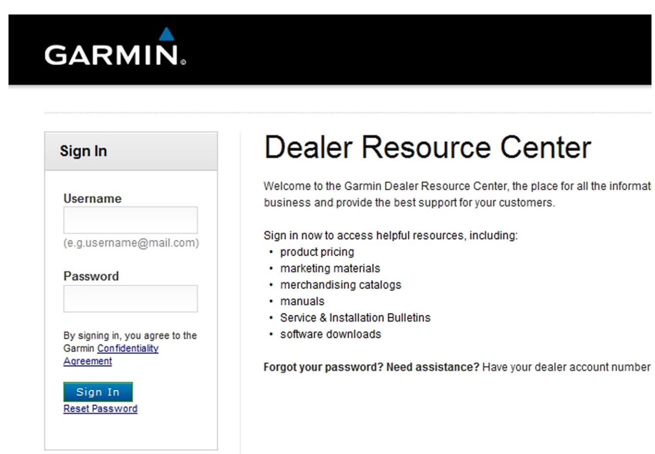
The following procedure requires a PC with an SD card reader. Piper recommends SanDisk® SDDR-99 and SDDR-93 card readers. Additionally, only SanDisk brand 16GB SD cards are recommended.

The software image is an executable self-extracting file which builds the correct file structure onto an SD card for use loading software to the G1000 NXi and GFC700. To obtain the current file follow the procedures outlined below.

NOTE: In order to create a loader card, the installer completing these procedures must be an authorized Piper service center to gain access to the necessary data via the Garmin website.

(a) Go to http://www.garmin.com and click on the 'Dealer Resource Center' link in the lower right hand portion of the home page. Enter username and password. See "Figure 40".

(b) Select Support and then Software Downloads.



Dealer Login Screen
Figure 40

(c) Search for the appropriate image version (i.e., 2164.xx from the appropriate "Chart 41" on page 342502227) in the “Search by Keyword” field.

(d) Save the file to a known location on the PC.

(e) Run the executable file that was downloaded. A window will appear to confirm software installation.

(f) Insert the SD card into the SD card reader and select the Next button in the pop-up window.

(g) In the pop-up window, select the drive that corresponds to the inserted SD card and then click the Next button.

NOTE: The selected drive will be completely erased. Click the Next button to acknowledge any warnings that appear. Another pop-up window will appear to monitor loader card creation progress. After the card creation is successfully completed, another pop-up window will appear.

(h) Select the Finish button.

(i) Stop the card reader using Windows stop device function.

(j) Eject the SD card from the card reader.

(k) Apply a label to the software loader card identifying the aircraft, G1000 NXi SFTW and the appropriate software image part number as shown below:

NOTE: See appropriate "Chart 41" on page 342502227 for software image part number.

M350
G1000 NXi SFTW
006-x2164-xx

(3) Handling a Software Download Card

The SD memory cards used for the Software Download Card are manufactured to the specifications of the SD Card Association for use in portable devices. As such they are reasonably robust devices that do not require special handling instructions. However, the memory cards are not unbreakable and reasonable precautions regarding their use and handling should be used including the following guidelines:

(a) Do no bend an SD memory card or subject it to violent impact.

(b) Do not leave SD cards in locations where they may be exposed to moisture or high temperatures.

(c) Avoid exposing the card to static electricity or electro-magnetic interference.

J. Software Files

(1) Definition

Software files are defined by part number and version number. Each G1000 NXi / GFC 700 LRU reports the software version it currently contains to the user in two places.

(a) Normal System Mode: The AUX – SYSTEM STATUS page lists each LRU and the reported software version.

(b) Configuration Mode: The SYSTEM STATUS page (SYSTEM page group) reports more detailed LRU information, including software version, part number, and LRU status. Software files are loaded to LRUs from the SYSTEM UPLOAD page in configuration mode.

(2) Troubleshooting

See "Chart 32".

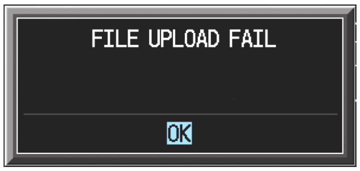
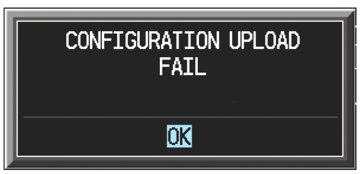
(3) Configuration File Descriptions

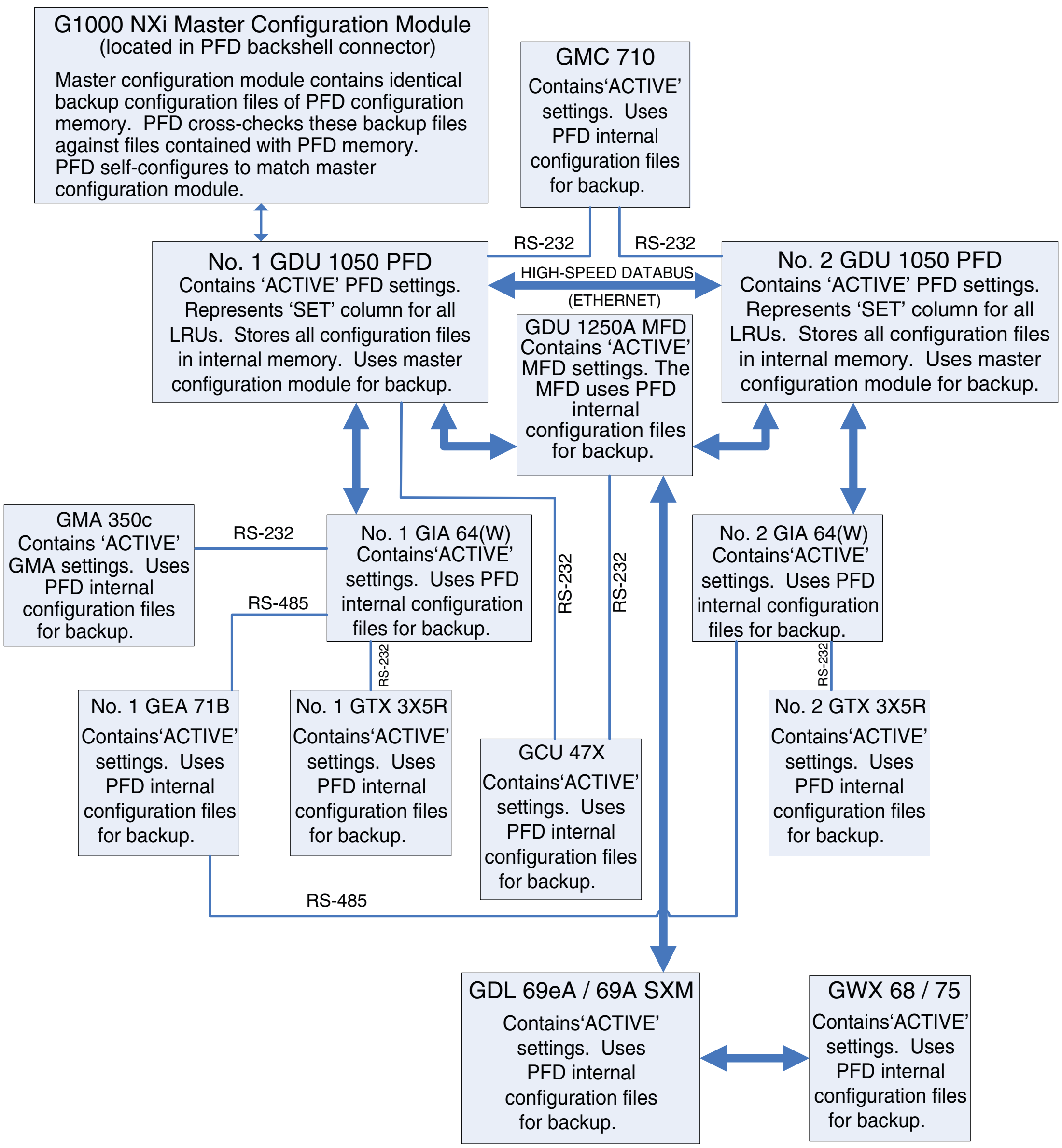
(a) Configuration Files are divided into groups, and are only displayed at the System Upload page during the software/configuration loading process.

(b) Configuration files contain preset selections for input/output channels, aircraft-specific settings, and LRU-specific settings. The following list describes each of the configuration files:

AIRFRAME	This file contains data such as airspeed parameters, engine/airframe sensor limitations, fuel tank parameters and alerting system settings that tailor a PFD or MFD to the PA-46-350P.
ALERTS	Specific alerts are set when this file is loaded.
SYSTEM	This file configures the G1000 NXi high-speed data bus (HSDB) to expect two PFDs, MFD, two GIAs, and a GDL69eA.
MANIFEST	This file loads a manifest checklist of all software part numbers and versions associated with an approved system configuration. The G1000 NXi performs a software check between each LRU's reported version and the version contained on the manifest. If an inequality is detected for an LRU, this LRU is then excluded from the G1000 NXi and a manifest alert is triggered to the operator.
MFD1	This file configures MFD serial/discrete communication and alert system settings.
PFD1/PFD2	These files configure PFD1/PFD2 serial/discrete communication and alert system settings.
GCU	This file configures GCU 47X serial/discrete communication settings.
GMC	This file configures GMC 710 serial/discrete communication settings.
GIA1/GIA2	These files configure GIA1/GIA2 serial/discrete communication settings.
GMA_PIL/	This file configures GMA audio and serial communication settings.
GTX1/GTX2	These files configure GTX 1/GTX 2 transponder and serial communications settings.
GEA	These files configure GEA engine/airframe parameters.
GDC_PIL/ GDC_COPIL	This file configures GDC 72 air data values.
GDL_69	This file configures gains and cable loss.
GWX	This file configures parameters for the radar installation.
AUDIO	This file configures audio alerts.

CHART 32
G1000 NXi - TROUBLESHOOTING / MESSAGE ADVISORIES - SOFTWARE CONFIGURATION

Problem	Solutions
MFD or PFD displays do not power up:	• Ensure power is present at display backshell connector. • Troubleshoot per the “Blank Display” GDU section.
Software file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry software file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with software version and Card Loader. • Replace LRU.
Configuration file load fails: 	• Ensure that LRU is reporting data on System Status page (LRU is ‘ONLINE’). Check data path wiring as needed. • Retry configuration file load or try using a different card. Ensure that the MFD is not touched during the loading process. • Ensure that LRU part number is compatible with Card Loader. • Replace LRU.
GIA1 and/or GIA2 to ‘LRU’ data path not working:	• Ensure GIA1 and GIA2 are configured correctly. • Check wiring, connectors & pins as needed.
Software File Mismatch Alert appears in lower right corner of PFD when started in normal mode: 	• Ensure that proper software file part number and version were loaded to LRU. See the appropriate Required Equipment List (REL), "Chart 41" on page 342502227. • Check and ensure that correct Loader Card was used during load process. See "G1000 NXi / GFC 700 Software Information" above. • Reload software to LRU.



LRU Configuration File Storage
Figure 41

(4) Configuration File Storage

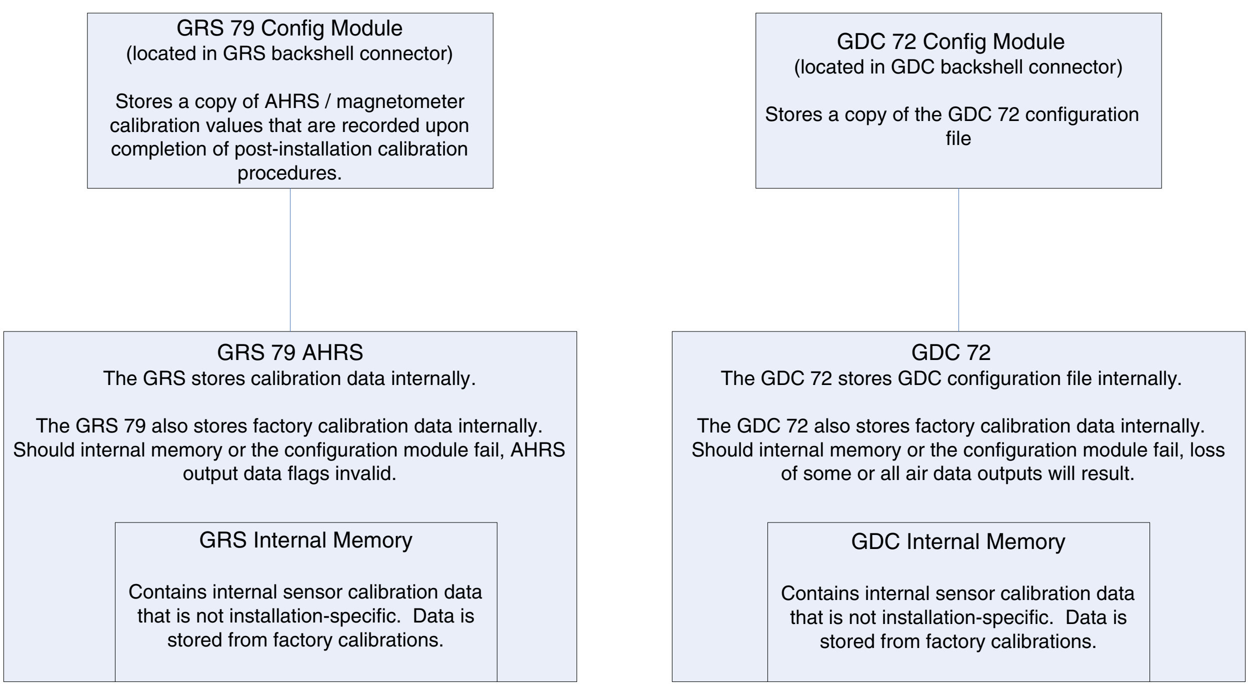
The G1000 NXi system is designed to store all configuration settings in various places so that the configuration is retained in the aircraft during maintenance of units.

During system configuration, as shown in "Figure 41", each file is sent directly to the applicable LRU where it is stored in local LRU memory (except GRS 79 & GDC 72). Each file is also stored in the PFD internal memory. The applicable PFD also sends a copy of all configuration files to the 'Master Configuration module', located in the connector backshell (see "Figure 33" on page 342502130). If the PFD is replaced, the configuration module retains all configuration files in the aircraft.

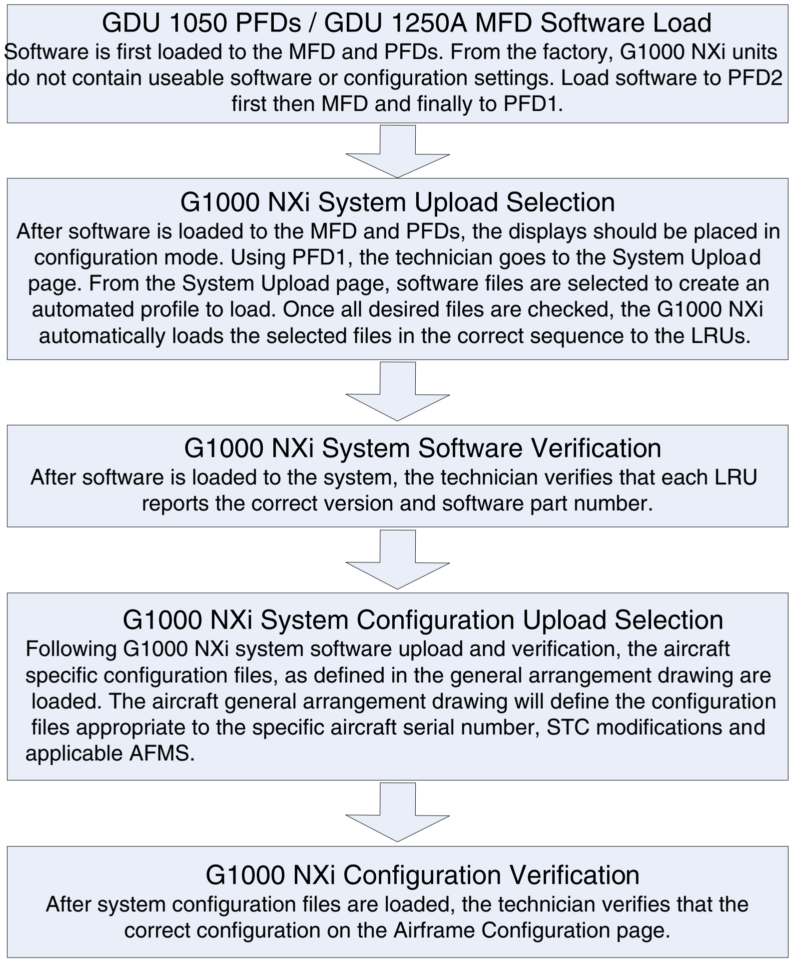
NOTE: The GRS 79 AHRS and GMU 44(B) Magnetometer do not have a configuration file. However, these LRUs do store calibration data acquired during the post installation checkout, which are characteristic to the specific installation. While performing maintenance on these units, re-calibration may be required. See GRS 79 AHRS / GMU 44(B) Magnetometer for more information on re-calibration criteria.

The GRS 79 and GDC 72 configuration modules, as shown in "Figure 42", function differently than the rest of the system. The GDC 72's configuration file is loaded directly to GDC internal memory. A copy of the file is stored in the GDC configuration module.

The GRS 79 configuration module does not store any configuration settings. Instead, it stores calibration data recorded during installation calibration procedures.



GRS/GDC Configuration Settings Storage
Figure 42



Software/Configuration Overview
Figure 43

(5) G1000 NXi Software/Configuration Procedure

See "Chart 33".

This section summarizes the procedures required to load software and configuration files to the G1000 NXi. It is intended to work as a central guide for technicians to use while performing maintenance on the aircraft. In sections of this manual where software is required to be reloaded, these sections will make reference back to this Section for instructions. The technician should use proper judgment regarding the context of maintenance required while following this section.

The diagram in "Figure 43" depicts an overview of the software/configuration sequence for the G1000 NXi system. This applies mostly to a new G1000 NXi system which has not previously been powered up and is for informative purposes only.

CHART 33
G1000 NXi - SYSTEM COMMUNICATION HIERARCHY

The following criteria must be satisfied to be able to perform the desired operation:	
Desired Operation	**Criteria for Success**
Load Software to MFD or PFD Displays	• G1000 NXi Loader Card must be inserted in top slot for each display to be loaded. • CLR & ENT keys must be held during power up of display. • Power on only one display at a time during software loading.
Load AIRFRAME, SYSTEM, MFD, PFD1, PFD2 and MANIFEST configuration files to MFD and PFDs	• G1000 NXi Loader Card must be inserted in top slot of PFD1. • PFD1 and MFD must be powered on. • PFD1 and MFD must have correct software.
Load Software/Configuration files to GIA 64(W)s	• G1000 NXi Loader Card must be inserted in top slot of PFD1. • G1000 NXi system must be powered on. • PFD and MFD must have correct software. • PFD 1 and MFD must be successfully configured with AIRFRAME, SYSTEM, MANIFEST, MFD, PFD1 and PFD2 configuration files.
Load Software/Configuration files to: - GMA 350c - GDC 72 - GEA 71B - GRS 79 (software only) - GMU 44(B) (software only) - GTX 3X5R	• G1000 NXi Loader Card must be inserted into PFD1 top slot. • G1000 NXi must be powered on. • PFD1 and MFD must have correct software and configuration settings. • GIA 64(W)s must have correct software. • GIA 64(W)s must be successfully configured with GIA1 and GIA2 configuration files. • Data path from GIA1 to each LRU must be operational.

K. LRU Replacement

Whenever a Garmin LRU has been replaced and the system is then booted into configuration mode, the Configuration Manager page appears and shows the synchronization status for the system and will note LRUs needing updates to software and/or configuration.

If the technician skips synchronization of the physically replaced LRUs then a “LRU replacement detected” message will be shown on the GDU in normal mode. This message will persist until the "Intelligent LRU Replacement Procedure" or the manual "Complete Software Load" on page 342502157 is performed. The “LRU replacement detected” message may not be displayed if there are existing Manifest error or Configuration error messages.

Intelligent LRU Software Replacement or manual Complete Software Load is the required action to repair messages related to Manifest error, Configuration error or LRU replacement detection that show up after physical replacement of an LRU.

(1) Applicability and Restrictions

(a) The Intelligent/Manual LRU Replacement procedure is not required if GDUs within the system are swapped in position. E.g., PFD display swapped with MFD display. The configuration information is stored by all GDUs in their internal memory. The information inside PFD harness configuration module such as fuel, cal and other data is retained during PFD replacement.

(b) The system keeps track of features that have been activated using enablement cards. If an LRU is replaced, Intelligent LRU Replacement will ensure that associated feature set remains available.

(c) Currently Intelligent LRU Replacement only supports replacement of one LRU at a time.

(d) Intelligent replacement only supports replacement of LRUs with the same hardware model number E.g., GIA 63W-20 should only be replaced with GIA 63W-20 model in order to utilize intelligent replacement. The manual "Complete Software Load" on page 342502157 can be used if the LRU hardware model is different. Intelligent LRU Replacement cannot be used if a software update is also being carried out for the system. Such an update will require a new loader card and must be carried out via the manual Complete Software Load using the System Upload page.

(e) Intelligent LRU Replacement can be used to reload any item that has been setup via a loader or enablement card. This includes LRU software, LRU configuration, equipment options and feature unlock settings.

(2) Transaction Log

In order for the G1000 NXi system to accurately replace an LRU’s software and configuration, it must first know what was originally loaded to that LRU. To facilitate this, the system records all updates performed through a loader card. Once this record is stored; the system will have the ability to replace those updates back to the LRU at any time. This stored record of all software and configuration updates is known as the “Transaction Log”.

The Transaction Log is viewable on the GDU in configuration mode via the Configuration Log page. The history can also be exported to the Top Slot SD card in a textual format by pressing the “DNLD TRANS” button on that page.

On the Configuration Log page the most recent update is at the top of the list. Grayed-out text represents configuration items that were loaded prior to the last “Clear Command”. These configurations are no longer applicable to the current state of the aircraft; but are listed in the log for informational and historical purposes.

Garmin's calibration and configuration features use specific files known as GREG files to upload calibration and configuration features. Each file is populated with various product data that is specific to a fleet of aircraft or may be specific to a single aircraft. As part of the transaction functionality, files that are used to load configuration and calibration data will be put through CRC generating software to generate a unique value for each transaction. Calibration and configuration data that is manually changed by maintenance technicians will not be recorded by the transactions logs. This includes calibration data changed via configuration pages, rigging, etc., however, transactions for the clear command softkeys that are part of the Systems Upload page will be recorded.

(3) Card Copy Feature

The system provides an option to copy loader cards to GDU internal memory. The internally saved loader cards are used by Intelligent LRU Replacement procedure. The configuration mode Card Copy option uploads the loader card into PFD internal memory. This process requires approximately four (4) minutes. A newly uploaded loader card is automatically synchronized with the MFD by transferring it in the background during normal mode. This transfer takes a similar amount of time.

The GDUs can hold copies of 32 cards. An interface is provided to delete cards that are no longer applicable. This can be accessed by using the "MANAGE" key on LRU Replacement page.

(4) Intelligent LRU Replacement Procedure

The intelligent LRU replacement procedure is performed on the LRU Replacement page available in configuration mode. Proceed as follows:

(a) Summary

1) Check the synchronization status of LRUs on the Configuration Manager page in configuration mode. This page lists those LRUs requiring synchronization.

2) Navigate to the LRU Replacement page, for the replaced LRU (shown in yellow text). Load the LRU configuration from the menu. No loader card or selection of options are required.

3) Verify the status of synchronization via the Configuration Manager page. A fully synchronized system will show "Synchronized" for all LRUs.

(b) Details

1) Selection of the replaced LRU.

The page will highlight LRUs that the G1000 NXi system recognizes as being "new" to the system. The system will provide a "LRU Replacement Detected" message by comparing serial numbers of the units. Serial number comparison is carried out for software version and region configuration version. Other configuration items are checked by comparing CRCs.

Some replaced LRUs may not get this message if they do not report a serial number. This behavior is common for GIA COM and GIA NAV items because they do not report a serial number. However, if these components have a different software or configuration version, it will be indicated by error messages for Manifest or Configuration. That will be an indication that a Manual/Intelligent Replacement procedure for software/configuration is required.

On the LRU replacement page, LRUs are highlighted (in yellow) if the software, configuration, or serial number of the new unit is different than the unit it is replacing.

2) Summary and control of what software/configuration will be loaded to the newly replaced LRU.

The control and display of this information is similar to the System Upload page.

3) Initiate the upload of software and configuration to the replaced LRU.

The page will report status and progress similar to the System Upload page.

4) Verification

Once an LRU Replacement procedure has been performed, installers can confirm that the system has been successfully returned to its previously approved state by using the Configuration Manager page available in configuration mode. This page supports to two functions:

a) Display of expected and actual G1000 NXi fleet and aircraft identification numbers.

b) Display of the synchronization status between each LRU and the GDU currently being used. This status compares the GDU's expected configuration to what each LRU is reporting as its actual configuration. If they match, the page will report "Synchronized". If they do not match, the page will report "Not Synchronized".

During the time between when the LRU hardware has been replaced, and prior to the LRU Replacement procedure being performed technicians should expect to see a "Not Synchronized" status. Once the LRU Replacement procedure is performed the status will return to "Synchronized", confirming that the replacement was successful

Display of "Invalid"/"Not Synchronized" messages after a replacement procedure indicate that a system restart is required. The restart will resolve items responsible for these error messages.

(c) Step-by-Step Procedure

The following assumes a new LRU has not yet been installed. Please make sure power is OFF before beginning.

NOTE: Only one LRU can be replaced at a time using this loading method.

1) Replace faulty LRU with new unit in accordance with that specific LRU's installation instructions. Ensure that Power is OFF before you begin.
2) Power up the PFD and MFD in config mode by holding down the ENT key until "INITIALIZING SYSTEM" appears on each GDU.
3) On the PFD, use the FMS knob to navigate to the "LRU REPLACEMENT" page.
4) The system should automatically detect the new LRU (displayed in gold letters). Use the FMS knob to select the desired LRU.
5) Ensure all SOFTWARE and CONFIGURATION boxes are checked.
6) Press the LOAD softkey.
7) Review the LRU replacement page to ensure that all SOFTWARE and CONFIGURATION items are marked as "PASS".
8) Remove power from the PFD and MFD.
9) Restart the PFD and MFD in config mode.
10) Use the FMS knob on the PFD to navigate to the "CONFIGURATION MANAGER" page.
11) Ensure that all CONFIGURATION ITEMS are marked as "Synchronized".
12) Use the FMS knob on the PFD to navigate to the "SYSTEM STATUS" page.
13) Use the FMS knob to scroll down to the recently installed LRU and confirm that the status is OK.
14) Remove power from the PFD and MFD.
15) Restart the PFD and MFD in normal mode.
16) Confirm that no MANIFEST, CONFIG ERROR, or LRU REPLACEMENT messages exist.
17) Review the Return-to-Service criteria for specific LRU under "Components" on page 34250267.

L. Complete Software Load

(PIR-107998 G.)

(1) PFD Software Load

CAUTION: DO NOT INTERRUPT POWER TO THE AIRCRAFT AT ANY TIME DURING THESE PROCEDURES. INTERRUPTION OF POWER COULD RESULT IN DAMAGE TO THE EQUIPMENT.

(a) Apply power to the aircraft and all avionics.

(b) Pull the MFD and both PFD circuit breakers. Remove any SD cards that were previously inserted in either PFD or MFD.

(c) Insert G1000 NXi/PA-46 M350 Loader Card into the PFD1 top card slot.

(d) While holding the ENT key on the PFD1, restore power to the PFD1 by pushing in the PFD1 circuit breaker.

(e) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFD1, release the ENT key.

(f) Press the YES softkey at the following prompt:

“DO YOU WANT TO UPDATE SYSTEM FILES?”

“NO WILL BE ASSUMED IN 30 SECONDS”

“UPDATING SYSTEM FILES, PLEASE WAIT.”

(g) Press the YES softkey at the following prompt (prompt may not appear):

“DO YOU WANT TO UPDATE SPLASH SCREEN?” (i.e., Custom Graphics Files)

“NO WILL BE ASSUMED IN 30 SECONDS”

(h) Press any softkey at the following prompt to confirm load completion:

“PRESS ANY KEY TO CONTINUE.”

“CONTINUING IN 10 SECONDS.”

(i) When PFD1 update is complete, it starts in the configuration mode. DO NOT remove power.

(j) While holding the ENT key on the MFD, restore power to the MFD.

(k) While holding the ENT key on the #2 PFD, restore power to the #2 PFD.

CAUTION: FOR THE REST OF THE SOFTWARE/CONFIGURATION PROCEDURE, DO NOT OPERATE THE MFD OR PFD2 WHILE LOADING SOFTWARE OR CONFIGURATION FILES UNLESS SPECIFICALLY INSTRUCTED TO DO SO. A FAILED OR CANCELLED LOAD MAY RESULT.

(2) System Upload

NOTE: To Select/Deselect the cursor in fields, press the small FMS knob on the PFD. The large FMS knob changes Groups, and the small FMS knob changes Pages within a Group.

(a) Go to the SYSTEM UPLOAD page on PFD1 using the small FMS knob.

1) Activate the cursor, select the GROUP field, and rotate the small FMS knob to generate a pick list. Rotate the small FMS knob to select Baseline.

2) Press ENT.

3) In the ITEM field, rotate the small FMS knob to generate a pick list and select the appropriate item (i.e., Piper PA-46-350P Mirage).

(b) Press ENT.

(c) Press the LOAD softkey (The load takes approximately 1.5–2 hours to complete).

(d) Press ENT to accept the software load.

(e) Deselect cursor.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the loader card from top slot of PFD1.

(3) Optional Equipment Configuration

Only the options listed below are approved for installation.

NOTE: Use "Chart 34" to determine which GTX and GTS options to install based on the aircraft configuration. If Flight Stream 510 is installed, also install the Flight Stream 510 option.

Perform the following steps if any of the following options are installed:

ADF RA-3504	Flight Stream 510
De-ice Installation	KN 63 DME
Enhanced AFCS Support	ARTEX ELT
GDL 69 SXM	GSR 56 Iridium
GSR 56 Voice/Text Only	No. 1 GTX 335R w/ GTS
No. 1 GTX 335R	No. 1 GTX 33D ES
Enable TIS-A for GTX 335R	Enable TIS-A for GTX 33D
No. 1 GTX 33D ES w/ GTS	No. 2 GTX 345R w/ GTS
No. 2 GTX 345R	No. 2 GTX 335R
GWX 68 / 75	Speedbrake Installation
WX-500 Storm Scope	GMU 44(B)
Battery Off	True Blue Inverter

(a) On PFD1, select the System Upload page and activate cursor.

(b) In the GROUP field, create a pick list by rotating the small FMS knob and select Options (Common), Options (Configuration A), or Options (Configuration B).

(c) Press ENT.

(d) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired option, press ENT.

(e) Verify both Software and Configuration boxes are checked for Garmin options.

CHART 34
TRANSPONDER CONFIGURATION OPTIONS

CONFIGURATION A (GTX 335 or GTX 33D in #1 GTX position)				
#1 Transponder	**#2 Transponder**	**GTS**	**FS 510**	**Options to Load**
GTX 335	None	No	Either	No. 1 GTX 335R & Enable TIS-A for GTX 335R (no GTS 825, no GTX 345R)
GTX 335	None	Yes	Either	No. 1 GTX 335R with GTS (no GTX 345R)
GTX 335	GTX 345	No	No	No. 1 GTX 335R & No. 2 GTX 345R (no GTS 825)
GTX 335	GTX 345	No	Yes	No. 1 GTX 335R & No. 2 GTX 345R without GTS and with Flight Stream
GTX 335	GTX 345	Yes	No	No. 1 GTX 335R & No. 2 GTX 345R with GTS
GTX 335	GTX 345	Yes	Yes	No. 1 GTX 335R & No. 2 GTX 345R with GTS and Flight Stream
GTX 33D	None	No	Either	No. 1 GTX 33D & Enable TIS-A for GTX 33D (no GTS 825, no GTX 345R)
GTX 33D	None	Yes	Either	No.1 GTX 33D ES with GTS (no GTX 345R)
GTX 33D	GTX 345	No	No	No. 1 GTX 33D ES & No. 2 GTX 345R (no GTS 825)
GTX 33D	GTX 345	No	Yes	No. 1 GTX 33D ES & No. 2 GTX 345R without GTS and with Flight Stream
GTX 33D	GTX 345	Yes	No	No. 1 GTX 33D ES & No. 2 GTX 345R with GTS
GTX 33D	GTX 345	Yes	Yes	No. 1 GTX 33D ES & No. 2 GTX 345R with GTS and Flight Stream

CONFIGURATION B (GTX 345 or GTX 345DR in #1 GTX position)			
#1 Transponder	**GTS**	**FS 510**	**Options to Load**
GTX 345	No	No	No. 1 GTX 345 (no GTS 825)
GTX 345	No	Yes	No. 1 GTX 345 without GTS and with Flight Stream
GTX 345	Yes	No	No. 1 GTX 345 with GTS
GTX 345	Yes	Yes	No. 1 GTX 345 with GTS and Flight Stream
GTX 345DR	No	No	No. 1 GTX 345DR (no GTS 825)
GTX 345DR	No	Yes	No. 1 GTX 345DR without GTS and with Flight Stream
GTX 345DR	Yes	No	No. 1 GTX 345DR with GTS
GTX 345DR	Yes	Yes	No. 1 GTX 345DR with GTS and Flight Stream

(f) Verify Configuration boxes checked for all other options.

(g) Press LOAD softkey.

(h) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

(i) Press ENT to accept.

(j) For other installed options, repeat steps (d) thru (i) by rotating the large FMS knob and highlighting desired option in the ITEM Field.

(k) De-activate the cursor.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the loader card from top slot of PFD1.

(4) Calibration Option Load

NOTE: Only load configuration option once. Re-loading of configuration option could result in a loss of system calibrations.

(a) Go to the System Upload page and activate cursor.

1) In the GROUP field, create a pick list by rotating the small FMS knob and select Options.

2) Press ENT.

3) In the ITEM field, create a pick list by rotating the small FMS knob and select the desired calibration (Flaps Calibration Initialization or Fuel Calibration Initialization).

4) Press LOAD softkey.

5) Verify the summary field lists the software and configuration are complete as required and PASS is displayed next to both appropriate boxes.

6) Press UPDT CFG softkey.

7) Press ENT to select YES when prompted “Update Config Module?”

(b) Press ENT to confirm completion.

NOTE: Upon completion of all required baseline config, option config, and calibration loading procedures, remove the loader card from top slot of PFD1.

(5) Splash Screen Update

(a) Remove power from PFD1, PFD2, and MFD.

(b) Insert the Loader Card into the PFD2 top card slot.

(c) Apply power to PFD2.

(d) Press the NO softkey at the following prompt:

"DO YOU WANT TO UPDATE SYSTEM FILES?"

(e) Press the YES softkey at the following prompt:

"DO YOU WANT TO UPDATE SPLASH SCREEN?" (i.e., Custom Graphics Files)

(f) At the prompt, press any key to continue.

(g) Remove the Loader Card from the PFD2 top card slot.

(h) Repeat steps (b) thru (g) for MFD, and then PFD1.

(i) Upon completion of splash screen update, remove the Loader card from top slot of PFD1.

(6) Software Load Confirmation

(a) Each PFD and MFD running in configuration mode.

(b) On PFD1, select the Systems Status page. Activate the cursor and toggle to the LRU window.

(c) Highlight each of the listed items in the LRU window and check the software part number and software version against the current numbers and versions shown in the appropriate Required Equipment List (REL), "Chart 41" on page 342502227. Use "Chart 40" on page 342502226 to record. Some items may be within the expandable list (denoted by a "+" next to the LRU) of the corresponding item. To expand the list, press ENT with the desired LRU highlighted

NOTE: Not all of the software P/Ns and versions on the Systems Status page need to be verified, only those listed in the appropriate REL.

(d) De-activate the cursor.

NOTE: If any software version or part number does not match, or is not successfully loaded, do not continue with post installation procedures. Troubleshoot and resolve the issue before continuing.

(e) Re-start PFD1, PFD2 and MFD.

(f) On MFD splash screen, verify appropriate version number ("PA-46-350P System xxxx.xx") is displayed in the upper right corner of the MFD.

(g) Select softkey on MFD to proceed beyond splash screen.

(h) Verify that no manifest (software mismatch) alerts are shown on any of the displays.

M. Configuration and Setup

Start #1 PFD, MFD, and #2 PFD in configuration mode by holding ENT key while applying power.

(1) XM Radio (GDL 69eA / 69A SXM)

See "Chart 25" on page 342502115, if the XM Satellite Radio is installed and requires activation.

(2) Aircraft Registration Number Entry (Transponder Configuration)

NOTE: The large FMS knob changes Groups and the small FMS knob changes Pages within a Group.

1) Place the PA-46-350P G1000 NXi loader card in the top slot of the #1 PFD.

2) Start the #1 PFD, MFD, and #2 PFD in the configuration mode by holding the ENT key while applying power.

3) Select the SYSTEM group by rotating the large FMS knob, and then select the AIRCRAFT CONFIGURATION page by rotating the small FMS knob.
4) Activate the cursor and highlight the AIRCRAFT REGISTRATION field. Use the small/large FMS knob to enter the aircraft registration number. Then press ENT key.
5) Ensure that an ICAO ADDRESS is automatically populated.
6) Press the SET GTX1 softkey to configure the transponder. Press ENT to select OK at the dialog that pops up. Then press ENT again after configuration is completed.
7) If dual GTX are installed, press the SET GTX2 softkey. Press ENT to select OK at the dialog that pops up. Then press ENT again after configuration is completed.
8) If GTS 825 is installed, press SET GTS softkey to configure traffic system. Press ENT to select OK at the dialog that pops up. Then press ENT again after configuration is completed.
9) Select the GTX group by rotating the large FMS knob, and then select the Transponder Airframe Configuration page by rotating the small FMS knob.
10) Activate the cursor and highlight the Allow Flight ID Entry field within the Flight ID Options window, select “Same as Tail” by rotating the small FMS knob, press ENT key.
11) Press “Set>GTX1” softkey to confirm the setting for #1 transponder. Wait for the configuration to complete.
12) Verify that the Default Flight ID now displays the entered Aircraft Registration Number in the XPDR 1 active column. If it doesn’t, repeat step 11).
13) Press “Set>GTX2” softkey to confirm the setting for #2 transponder (if installed). Wait for the configuration to complete.
14) Verify that the Default Flight ID now displays the entered Aircraft Registration Number in the XPDR 2 active column (if installed). If it doesn’t, repeat step 13).
15) Select the SYSTEM group by rotating the large FMS knob, and then select the Configuration Manager page by rotating the small FMS knob.
16) Press the CNFM CFG softkey and press ENT to select OK at the pop-up dialog.

(3) Flaps Calibration

NOTE: The large ‘FMS’ knob changes Groups and the small ‘FMS’ knob changes Pages within a Group.

(a) Start the #1 PFD, MFD, and #2 PFD in the configuration mode by holding the ENT key while applying power.
(b) Select the ‘CAL’ group, then select the ‘FLAPS AND TRIM CALIBRATION’ page on ‘PFD1.
(c) Unlock the Calibration page by entering the following softkey password:
   - 1 (far left softkey)
   - 2
   - 3
   - 4
(d) Press the ‘FLAP POSITION RESET’ SOFTKEY (3rd softkey). Press the ‘ENT’ key to acknowledge the prompt after the ‘RESET’ softkey is pressed.
(e) Put the flaps in the ‘FULL UP’ position.

(f) Press the ‘FLAP POSITION UP’ softkey (1st softkey).

(g) Put the flaps in the ‘FULL DOWN’ position.

(h) Press the ‘FLAP POSITION DN’ softkey (2nd softkey).

(i) Return the flaps to the ‘FULL UP’ position.

(j) Verify ‘CAL Value’ in the ‘FLAP POSITION’ window indicates 0.00 ± 0.02. If it does not, repeat steps (c) – (h) until it does.

(k) The calibration is complete.

(4) Iridium (GSR 56)

See "Chart 27" on page 342501119, if the Iridium Satellite Radio/Datalink is installed and requires activation.

(5) Option Unlock Procedures

WARNING: PRIOR TO ATTEMPTING UNLOCK OF ANY OF THE FOLLOWING OPTIONS, ENSURE THAT THE SOFTWARE IS PROPERLY LOADED AND THAT THE CONFIGURATION MODULE IS FUNCTIONAL. SEE "SOFTWARE LOAD" ON PAGE 342501152, TO VERIFY SOFTWARE HAS BEEN PROPERLY LOADED. IN CONFIG MODE ON THE PFD USING THE FMS KNOB GO TO GDU GROUP, KEY TEST PAGE AND VERIFY THAT “IICEEPROM” BOX IS CHECKED GREEN TO VERIFY THE CONFIGURATION MODULE IS FUNCTIONAL.

(a) CHARTVIEW Unlock (Optional)

1) Remove power from both PFDs and MFD by opening the PFD and MFD circuit breakers.

2) Insert the Chart Unlock card, in the upper slot of the PFD1.

NOTE: The Chartview Unlock card can only enable Charts on one system (one aircraft) and is locked to the system ID for that aircraft.

3) While holding the ENT key on both PFDs and MFD, restore power by closing the circuit breakers.

4) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the MFD and PFDs release the ENT key.

5) On PFD1, go to the System Upload page.

6) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.

7) Use the small FMS knob to create a pick list and select Enable Chartview in the ITEM field. Press ENT.

8) Verify there is a check mark in the box in the configuration column for Airframe.

9) Press the LOAD softkey.

10) Monitor the status of the upload. When the upload is finished, Complete and Pass is displayed. Press ENT.

11) De-activate the cursor.

12) Power down the system and remove the Chartview Unlock card from the PFD.

13) After the Chart function is unlocked, the card should be stored with the aircraft records.

(b) TAWS Unlock (Optional)

1) Remove power from both PFDs and MFD by opening the PFD and MFD circuit breakers.
2) Insert the TAWS Unlock card in the upper slot of the PFD1 .

NOTE: The TAWS Unlock card can only enable TAWS on one system (one aircraft) and is locked to the system ID for that aircraft.

3) While holding the ENT key on the both PFDs and MFD, restore power by closing the PFD and MFD circuit breakers.
4) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.
5) On the PFD1, go to the System Upload page.
6) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.
7) Use the small FMS knob to create a pick list and select Enable TAWS in the ITEM field. Press ENT.
8) Verify there is a check mark in the box in the configuration column for Airframe.
9) Press the LOAD softkey.
10) Monitor the status of the upload. When the upload is finished, Complete and Pass is displayed. Press ENT.
11) De-activate the cursor.
12) Power down the system and remove the TAWS Unlock card from the PFD.
13) After the TAWS function is unlocked, the card should be stored with the aircraft records.

(c) Synthetic Vision Unlock (Optional)

See "SVS/Pathways Activation" on page 342502170.

(d) Enhanced AFCS Unlock

1) Remove power from the MFD and PFDs by opening the PFD and MFD circuit breakers.
2) Insert the Enhanced AFCS Enable card in the upper slot of the #1 PFD.

NOTE: The Enhanced AFCS Enable card can only enable TAWS on one system (one aircraft) and is locked to the system ID for that aircraft.

3) While holding the ENT key on both PFDs and MFD, restore power by closing the PFD and MFD circuit breakers.
4) When the words “INITIALIZING SYSTEM” appear in the upper left corner of the PFDs and MFD release the ENT key.
5) On the #1 PFD, go to the SYSTEM UPLOAD page with the small FMS knob.
6) Activate the cursor and use the small FMS knob to create a pick list and select Configuration Files in the GROUP field. Press ENT.
7) Use the small FMS knob to create a pick list and select Enhanced AFCS in the ITEM field. Press ENT.
8) Press the LOAD softkey.
9) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.
10) De-activate the cursor.
11) Power down the system and remove the Enhanced AFCS Enable card from the PFD and store it with the aircraft records.

(6) SurfaceWatch Unlock (Optional)

(a) Remove power from the MFD and PFDs by opening the PFD and MFD circuit breakers.

NOTE: The SurfaceWatch Enable card can only enable SurfaceWatch on one system (one aircraft) and is locked to the system ID for that aircraft.

(b) Insert the SurfaceWatch Enable card in the upper slot of the #1 PFD.

(c) While holding the ENT key on both PFDs and MFD, restore power by closing the PFD and MFD circuit breakers.

(d) When the words "INITIALIZING SYSTEM" appear in the upper left corner of the PFDs and MFD release the ENT key.

(e) On the #1 PFD, go to the SYSTEM UPLOAD page with the small FMS knob.

(f) Activate the cursor and use the small FMS knob to create a pick list and select Enable_SurfaceWatch in the GROUP field. Press ENT.

(g) Use the small FMS knob to create a pick list and select Enable SurfaceWatch in the ITEM field. Press ENT.

(h) Verify there is a check mark in the box in the configuration column for Airframe.

(i) Press the LOAD softkey.

(j) Monitor the status of the upload. When the upload is Complete and Pass is displayed, press ENT.

(k) De-activate the cursor.

(l) Power down the system and remove the SurfaceWatch Enable card from the PFD.

(m) After SurfaceWatch function is unlocked, the enable card is to be stored with the aircraft paperwork.

(7) Ground Clutter Suppression & Turbulence Detection Unlock (Optional)

(a) Remove power from the #1 PFD by opening the PFD1 circuit breaker.

NOTE: The GCS & TD Enable card can only enable GCS & TD on one system (one aircraft). A new GCS & TD Enable card must be used for each aircraft.

(b) Insert the GCS & TD Enable card in the upper slot of the #1 PFD.

(c) While holding the ENT key on #1 PFD, restore power by closing the PFD1 circuit breaker.

(d) Activate the cursor and use the small FMS knob to create a pick list and select Enable_TD_GCS in the GROUP field. Press ENT.

(e) Use the small FMS knob to create a pick list and select Doppler, TD, GCS Enablement in the ITEM field. Press ENT.

(f) Verify there is a check mark in the box in the configuration column for Airframe.

(g) Press the LOAD softkey.

(h) Monitor the upload status. When the upload is Complete and Pass is displayed, press ENT.

(i) De-activate the cursor.

(j) Power down the system and remove the GCS & TD Enable card from the #1 PFD.

(k) After GCS & TD function is unlocked, the enable card is to be stored with the aircraft paperwork.

(8) Aviation Database Loading Procedures

(a) GARMIN GDU Database Card Loading Procedure

NOTE: Garmin Field Service Engineering, resident at Piper, will be responsible for supplying updates for the GDU 1XXX Terrain Data Base Card.

1) Remove power from the PFDs and MFD by pulling the PFD and MFD circuit breakers.
2) The Database Card supplied with the airplane may contain any or all of the following; depending on which options were seleted at delivery: SafeTaxi, Terrain 4.9 Arc Second, Airport Directory, Basemap, Obstacle DB2, FliteCharts, IFR/VFR Charts, and/or NAV Database.
3) Install Database Card in the bottom slot of the MFD.
4) Restore power to the PFDs and MFD.
5) The MFD will load the databases and automatically sync them to the PFDs. Status can be monitored on the Databases page of the Aux group on the MFD.
6) After database sync is completed, follow onscreen instructions to restart the avionics.
7) Remove the card from the MFD bottom slot and store with the aircraft records.

(b) JEPPS NAV Database and Charts Procedure

The data card delivered with the airplane contains the "G1000 Electronic Chart Service"". The databases are current at the time of delivery. Up-to-date databases are available from Jeppesen on a subscription basis.

JEPPS CHARTS Loading Procedure (If Chartview installed.)

a) Remove power from the MFD by pulling the MFD circuit breaker.
b) Install Jepps Charts data card in the bottom slot of the MFD.
c) Restore power to the MFD by closing the MFD circuit breaker.
d) Press any key to confirm the database update.
e) After the update completes, the MFD starts in normal mode.
f) Navigate to the Databases page of the Aux group on the MFD. Remove power from the aircraft when instructed to cycle power.
g) Remove the data card from bottom slot of the MFD and store with the aircraft records.
h) Reapply aircraft power and verify charts are updated on the MFD splash screen.

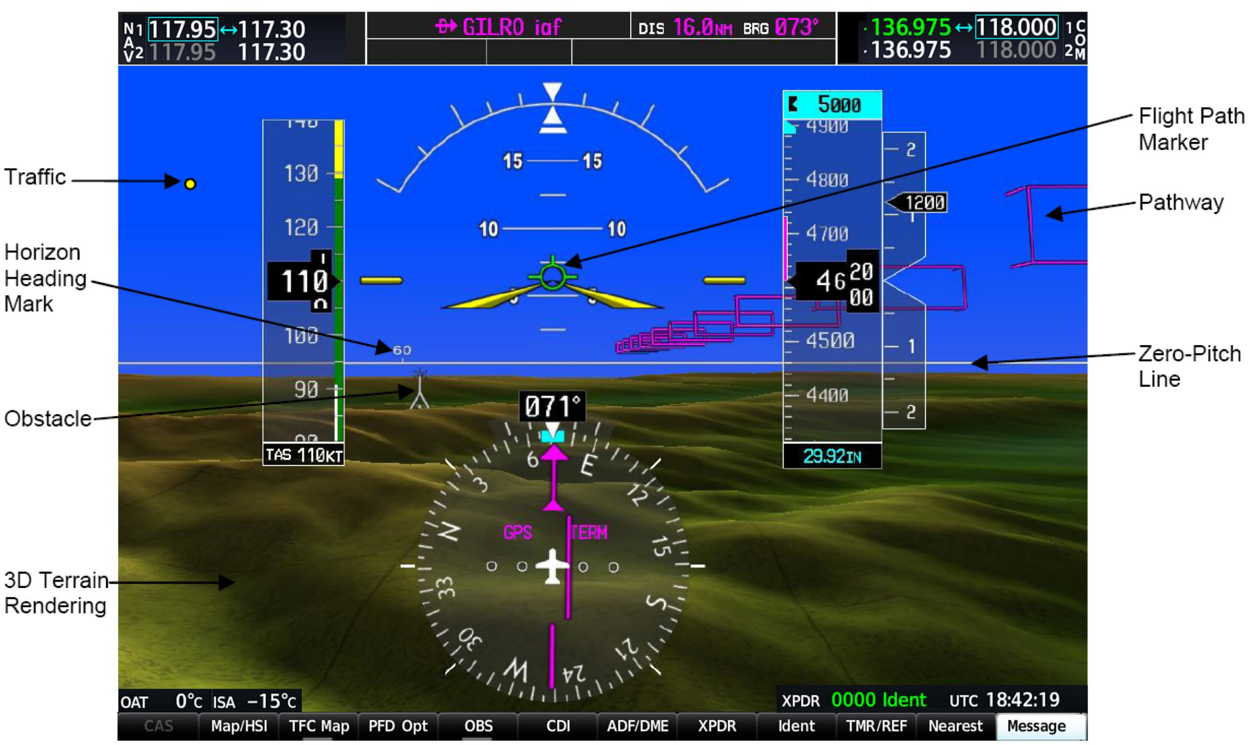
(9) Synthetic Vision System

(a) Description

The Synthetic Vision System (SVS) is intended to provide the crew with a greater awareness of the aircraft's position relative to surrounding terrain, obstacles, and traffic (optional). This is accomplished by placing a three dimensional depiction of terrain, obstacles, and traffic in the PFD primary field of view such that their proximity to the aircraft is more easily understood during instrument scanning. The display of SVS can be turned on and off by the pilot by a dedicated softkey on the PFD.

The SVS PFD frame-of-reference is aligned with the aircraft body frame, using AHRS attitude and heading data for orientation. Accordingly, the SVS is egocentric and attitude-aligned. GPS position and GPS-derived altitude are used for the SVS position and elevation references, respectively. The SVS 3D terrain presentation is generated from a high resolution 9 arc-second terrain database image. For the SVS system to be made available by the G1000 NXi, valid attitude, heading, GPS position, and terrain databases are required.

All previous existing PFD display features are retained and drawn over the SVS. For example, airspeed, altitude, vertical speed, heading, and HSI items will remain unchanged, and will be drawn over the SVS presentation.



SVS / Pathways Features
Figure 44

1) Synthetic Vision System

3D Terrain Rendering on PFD – SVS system will utilize an egocentric, forward looking perspective of 3D terrain rendering and shading, to be displayed on the PFD.

Zero Pitch Line – The PFD displays an artificial horizon pitch reference line above the display of terrain, similar to the existing artificial horizon which currently separates the blue/brown attitude depiction. The display of horizon heading marks in 30° intervals on the pitch line is a pilot-selectable feature.

Flight Path Marker – The projected aircraft flight path is displayed in the form of a Flight Path Marker on the PFD.

Display of Traffic – Traffic data from existing certified TIS, provided by the G1000 NXi GTX 335R transponder, is used to generate on-screen traffic symbols.

Display of Land-based Objects – Obstacles, runways, water bodies, and other land features are displayed on the PFD.

2) Pathways

SVS / Pathways are a pilot-selectable feature. The display of SVS can be turned on and off by the pilot by a dedicated softkey on the PFD. The display of Pathways requires that the 3D Terrain feature be displayed (i.e. SYN TERR softkey pressed).

When active, the Pathway is displayed on the PFD and will show flight path information in the form of a virtual 3D flight path, composed of rectangular elements which form a tunnel. The depiction of this pathway is derived from the existing FMS navigation capabilities of the G1000 NXi, and the display will be consistent with that of the active flight plan shown on the MFD.

3) Control & Operation

SVS/Pathways is an embedded software feature of the G1000 NXi and is controlled by using PFD softkeys. Specific details on operating the SVS/Pathways feature can be found in the Synthetic Vision & Pathways AFMS specified in the Pilot's Operating Handbook (POH).

4) Servicing

The software upgrades and SVS/Pathways function are non-serviceable software which is either active or inactive in the G1000 NXi system. Return to Service Procedure, below, provides a check to determine the state, and versions of software currently installed.

Reactivation may be required following normal maintenance procedures.

See "SVS/Pathways Activation" on page 342502170.

(b) Troubleshooting

The SVS/Pathways software feature requires the following G1000 NXi sensors/data to be valid:

1) AHRS
2) Heading
3) GPS Position
4) 9 Arc-Second Terrain Data

In the event that one the above items fails or is unavailable, the SVS/Pathways feature is automatically removed from the PFD. "Chart 35" describes possible symptoms associated with the SVS/Pathways feature, and provides corresponding actions for troubleshooting.

"Chart 36" on page 342502169 provides SVS/Pathways specific alert messages which may appear in the Alerts Window on the PFD (press the ALERTS softkey on the PFD to view the Alerts Window).

CHART 35
TROUBLESHOOTING - SYNTHETIC VISION SYSTEM

Symptom	Recommended Action
"SYN VIS" softkey does not appear on PFD softkey tier.	Follow steps in SVS/Pathways Activation, below, to reactivate the SVS/Pathways feature.
3D terrain presentation does not appear on PFD.	Verify that P/N 010-00330-43 terrain datacards are installed in the lower slot of the PFD and MFD. Verify that no alert messages are shown in the PFD Alerts Window. Verify that the G1000 NXi AHRS, and heading data are valid on the PFD. Verify that a valid GPS 3D position solution is being received. If a terrain database update has just been performed, allow the system time to initialize and verify the data. When the databases have been verified, the current database cycle and version are reported on the MFD AUX – System Status page.

CHART 36
SYNTHETIC VISION SYSTEM – ALERT MESSAGES

Failure Message	Cause	Solution
SVS – SVS DISABLED: Out of available terrain region.	SVS is disabled because the aircraft exceeded the boundaries of the loaded terrain database.	Geographical operation limitations are defined in the SVS/Pathways AFMS, specified in the POH.
SVS – SVS DISABLED: Terrain DB resolution too low.	SVS is disabled because a 9 Arc-Second or better database is not currently loaded.	Ensure P/N 010-00330-43 Terrain Cards are installed in the lower slot of each display. If terrain data has been recently updated, ensure that the correct 9 Arc-Second databases were used.

(c) Removal and Replacement

SVS/Pathways and all associated software version changes are non-serviceable software and cannot be physically removed or replaced in the G1000 NXi.

Reinstallation of the software upgrade and SVS reactivation should be accomplished as described below.

(d) Return to Service

1) SVS/Pathways Feature

Certain G1000 NXi maintenance activities will require the SVS/Pathways feature to be reactivated and/or the unit software to be reloaded.

Specifically, any time the default AIRFRAME configuration file is reloaded to the G1000 NXi, the SVS/Pathways feature 'activation' is lost. See "Configuration File Descriptions" on page 342502148, for detailed information regarding G1000 NXi configuration files.

The following specific maintenance scenarios will require re-activation of the SVS/Pathways feature:

a) Software and configuration files are completely re-loaded to the G1000 NXi system.

b) Software and configuration files specific to the PFD(s) are re-loaded, either as the result of troubleshooting procedures or complete PFD replacement.

In the event one of these scenarios occurs, the technician must reactivate and verify the SVS/Pathways feature following the steps outlined below.

2) Software Reloading

Removal and replacement of any LRU mandates the reloading of the applicable LRU software. Once completed, the software and configuration must be loaded into the new unit following the steps outlined in the configuration and software loading procedures (i.e., "G1000 NXi Software/Configuration Procedure" on page 342502152, "PFD Software Load" on page 342502157, "System Upload" on page 342502158, "Optional Equipment Configuration" on page 342502158, and "Software Load Confirmation" on page 342502161).

(e) SVS/Pathways Activation

This section provides the requirements and instructions necessary to unlock the G1000 NXi Synthetic Vision / Pathways feature.

1) Database Cards

The Garmin Synthetic Vision and Pathways feature requires 9 arc-second high resolution terrain databases to function.

Each G1000 NXi display must be equipped with the P/N 010-00330-43 Terrain/ Obstacle/ SafeTaxi database card installed in the lower slot.

2) SVS Activation Procedure

Activate the SVS/Pathways feature by performing the following steps:

a) Apply power to the G1000 NXi system.

b) Pull the MFD, PFD1, and PFD2 circuit breakers.

c) A special SVS Unlock card is required to activate this feature, see above. Insert this card in the upper slot of PFD1.

NOTE: The Synthetic Vision Unlock card can only enable Synthetic Vision on one system (one aircraft) and is locked to the system ID for that aircraft.

d) While holding the ENT key on the PFD1, PFD2, and MFD (for MFD press and hold the farthest right pushbutton), restore power to the displays.

e) When the words **INITIALIZING SYSTEM** appear in the upper left corner of the displays, release the ENT key.

f) On PFD1, go to the System Upload page using the FMS knob.

g) Activate the cursor. Use the small FMS knob to select CONFIGURATION FILES in the GROUP field and press the ENT key.

h) Highlight the ITEM field. Use the small FMS knob to select the “Enable SVS Dual PFD” option and press the ENT key.

i) Verify there is a check mark in the box in the configuration column for Airframe.

j) Press the LOAD softkey.

k) When the upload is Complete and Pass is displayed, press the ENT key.

l) De-activate the cursor.

m) Power down the system and remove the SVS Unlock card from the PFD.

n) After the SVS is unlocked, the Synthetic Vision Unlock card should be stored with the aircraft records.

3) SVS / Pathways Installation Verification

a) Apply power to the G1000 NXi system. Allow the AHRS and magnetometer systems to stabilize and align. Verify that air data information becomes valid on the PFDs. Check the MFD AUX – System Status page to verify GPS signals acquisition.

b) Press the ALERTS softkey on PFD1 and verify no database, manifest, or configuration errors exist.

c) Press the PFD softkey. Verify a SYN VIS softkey is shown in the lower left corner of the display.

d) Press the SYN VIS softkey, then press the SYN TERR softkey, to activate the Synthetic Vision terrain display feature. Verify that the traditional blue/brown attitude depiction is replaced with the Synthetic Vision rendering within 2-3 minutes of activation.

e) Installation is complete. Be sure to keep the SVS / Pathways unlock card with the aircraft for future use.

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N. Post-Installation Procedures

NOTE: This entire procedure must be successfully accomplished in order for the G1000 NXi system to be airworthy in your airplane.

(1) Bus Power Check

NOTE: The following checks MUST be performed prior to electrical power-up with the associated wiring connected to the electronic equipment. Failure to perform these checks prior to powering up may result in damage to the equipment.

(a) Do not apply external power to the aircraft.

(b) Verify that all circuit breakers are set.

(c) Turn ON Battery Master switch with Avionics Bus Master switch and the Emergency switch in the “OFF” position.

1) Verify Avionics Bus 1, Avionics Bus 2, and Emergency Bus are not powered.
2) Pull #1 Main Bus circuit breaker and Non-Essential Bus circuit breaker.
3) Verify Aft Main Bus and Forward Main Bus are powered.
4) Verify Non-Essential Bus is not powered.
5) Reset #1 Main Bus circuit breaker. Pull #2 Main Bus circuit breaker.
6) Verify Aft Main Bus and Forward Main Bus are powered.
7) Reset Non-Essential Bus circuit breaker. Pull #1 Main Bus circuit breaker.
8) Verify Aft Main Bus, Forward Main Bus, and Non-Essential Bus are powered.
9) Reset #1 Main Bus circuit breaker and #2 Main Bus circuit breaker.

(d) Turn ON the Avionics Bus Master switch.

1) Pull #1 Avionics Bus circuit breaker, #1 Main Bus circuit breaker, #2 Main Bus circuit breaker, and Non-Essential Bus circuit breaker.
2) Verify Avionics Bus 1 and Avionics Bus 2 are powered.
3) Verify Aft Main Bus, Forward Main Bus, Non-Essential Bus, and Emergency Bus are not powered.
4) Pull the Avionics Bus Tie circuit breaker.
5) Verify Avionics Bus 2 is powered and Avionics Bus 1 is not powered.
6) Reset #1 Avionics Bus circuit breaker. Pull #2 Avionics Bus circuit breaker.
7) Verify Avionics Bus 2 is not powered and Avionics Bus 1 is powered.
8) Reset #2 Avionics Bus circuit breaker, #1 Main Bus circuit breaker, #2 Main Bus circuit breaker, Non-Essential Bus circuit breaker, and Avionics Bus Tie circuit breaker.

(e) Turn ON the Emergency switch then place Battery Master switch and Avionics Bus Master Switch in the “OFF” position.

1) Verify that the Main Busses (Forward and Aft) and Non-Essential Bus are not powered.
2) Verify that the Avionics Busses (#1 and #2) are not powered.
3) Verify that the Emergency Bus is powered.

(f) Turn OFF the Emergency switch, Battery Master switch and Avionics Bus Master Switch.

Verify that the Battery Bus is powered.

(2) Wiring Harness Checkout

The following is required only when a harness has been repaired, modified, or replaced, but is recommended when troubleshooting.

Prior to installing any LRUs, the wiring harness should be checked for proper connections to the aircraft systems and other avionics systems. Point to point continuity should be checked to expose any faults such as shorting to ground. Any faults or discrepancies found should be corrected before proceeding. Refer to wiring schematics in Chapter 91.

(3) Initial Display Testing

Verify the following:

(a) COM/NAV fields are valid in the top corners of the PFDs .

(b) Altitude, airspeed, vertical speed, heading and OAT fields are valid on the PFDs.

(c) Verify the Aircraft Registration Number is displayed near the top right-hand corner of both PFDs.

1) If optional GTX2 is installed, perform steps 2) and 3).

2) Press XPDR softkey, then press XPDR1 or XPDR2 to toggle between #1 and #2 transponder.

3) Verify the Aircraft Registration Number is displayed near the top right-hand corner of both PFDs when XPDR1 is selected and when XPDR2 is selected.

(d) No BACKUP PATHs alerts on the PFDs. If an LRU is not communication over its primary path, the BACKUP PATH alert will identify which LRU is having the problem.

(e) Select Reversionary Mode on both displays by pressing the display backup switches (located above each display), and displays will display valid attitude, altitude, airspeed, vertical speed and engine instruments. De-activate the Reversionary Mode.

(f) Verify flap position indication matches actual flap position for 0°, 10°, 20°, and 36° ± 2°.

(g) DOOR AJAR annunciation:

1) Open cabin door.

2) Depress the door switches independently.

3) Verify “DOOR AJAR” annunciation does not clear from display.

4) Close door.

5) Verify “DOOR AJAR” annunciation is cleared from display.

6) Open baggage door.

7) Verify “DOOR AJAR” annunciation is displayed.

8) Close baggage door.

9) Verify “DOOR AJAR” annunciation is cleared from display.

(h) HYDR PUMP ON, GEAR DOWN, and VACUUM LOW discrete I/O’s.

1) Place the airplane on jacks per 7-10-00.

2) Put PFD1 in configuration mode by pressing the ENT key while applying power.

3) Using FMS knob on PFD1 go to GIA I/O Configuration page (GIA group, page 3).

4) Activate the cursor and using the large FMS knob scroll to channel “IN 16”.

5) Cycle the landing gear and verify the DATA box next to CHANNEL “IN 16” is green during operation of the hydraulic pump and black when the hydraulic pump is off (this verifies the HYDR PUMP ON discrete I/O is functioning).

6) Using the large FMS knob on the #1 PFD scroll to channel “IN* 1A”. Use large FMS knob to scroll the list to check items below as needed.

7) Verify the DATA boxes next to CHANNEL “IN* 1A”, “10A”, “14A”, and “17A” are green with the landing gear down and black with the landing gear up (this verifies the GEAR DOWN discrete I/O is functioning).

8) Verify the DATA boxes next to CHANNEL “IN* 2A”, “11A”, “15A”, and “16A” are black with the landing gear down and green with the landing gear up (this verifies the GEAR DOWN discrete I/O is functioning).

9) Using the large FMS knob on the #1 PFD scroll to channel “IN* 3”.

10) Verify the DATA box next to CHANNEL “IN* 3” is green with low vacuum (this verifies the VACUUM LOW on GIA1 and ICE FAIL on GIA2 discretes I/O are functioning).

11) Scroll to the top of the page and change the UNIT to GIA2. Repeat steps "4)" through "10)" to verify GIA2 discretes.

12) Cycle power to #1 PFD. Power-up #1 PFD in normal mode.

(i) STALL WARN FAIL annunciation.

1) Pull the “STALL WARN” circuit breaker.

2) Verify “STALL WARN FAIL” annunciation appears on display.

3) Close “STALL WARN” circuit breaker.

4) Verify “STALL WARN FAIL” annunciation is cleared from display.

(j) Pitot Heat annunciations.

1) Turn the Pitot Heat switch to the OFF position.

2) Verify “PITOT HEAT OFF” annunciation appears on display.

3) Turn the Pitot Heat switch to the ON position.

4) Verify “PITOT HEAT OFF” annunciation is cleared from display.

5) Pull the “L PITOT HEAT” circuit breaker.

6) Verify “L PITOT HT FAIL” annunciation appears on display.

7) Pull the “R PITOT HEAT” circuit breaker.

8) Verify “PITOT HEAT OFF” annunciation replaces “L PITOT HT FAIL” annunciation.

9) Close “L PITOT HEAT” circuit breaker.

10) Verify “R PITOT HT FAIL” annunciation replaces “PITOT HEAT OFF” annunciation.

11) Close “R PITOT HEAT” circuit breaker.

12) Verify “R PITOT HT FAIL” is cleared from display.

13) Turn OFF Prop Heat switch after test is complete.

(k) Avionics Cooling Fan Fail Annunciation Test

1) Pull the “MFD FAN” circuit breaker.
2) Verify “MFD FAN FAIL” advisory CAS on PFDs after an approximately 10 second delay.
3) Close the “MFD FAN” circuit breaker.
4) Verify “MFD FAN FAIL” advisory CAS is cleared from displays.
5) Pull the “PFD2 FAN” circuit breaker.
6) Verify “PFD2 FAN FAIL” advisory CAS on PFDs after an approximately 10 second delay.
7) Close the “PFD2 FAN” circuit breaker.
8) Verify “PFD2 FAN FAIL” advisory CAS is cleared from displays.
9) Pull the “PFD1 FAN” circuit breaker.
10) Verify “PFD1 FAN FAIL” advisory CAS on PFDs after an approximately 10 second delay.
11) Close the “PFD1 FAN” circuit breaker.
12) Verify “PFD1 FAN FAIL” message is cleared from display.
13) Pull the “AV FANS” circuit breaker.
14) Verify “AV1 FAN FAIL”, “AV2 FAN FAIL”, and "AV3 FAN FAIL" advisory CAS on PFDs after an approximately 10 second delay.
15) Close the “AV FANS” circuit breaker.
16) Verify FAN FAIL advisories are cleared.

(4) Emergency Bus Operation Verification

(a) Ensure Engine is “OFF”.
(b) Disconnect external power, if present.
(c) Turn Battery Master SWITCH “OFF“.
(d) Turn EMERGENCY switch “ON”.
(e) Verify all the following functions are operational:

1) PFD #1
2) COM #1
3) NAV #1
4) AHRS #1
5) Audio Panel
6) Air Data Computer #1
7) GEA (Engine Instruments)
8) Standby Instrument

NOTE: MFD, PFD2, AHRS2, GDL 69eA, Autopilot Controller, Transponder 1 and 2, Air Data Computer 2, Radar, Keypad, ADF, DME, GTS 825, and Stormscope are not available. Also note that NAV2, COM2, Oil Pressure, Manifold Pressure, Vacuum, Cabin Altitude, and Differential Pressure display invalid.

(5) G1000 NXi Checks

Perform the following steps to verify the following:

NOTE: Do not have External Power on the aircraft while conducting the following tests.

(a) AHRS and ADC data path check

1) In the GIA page group, go to the GIA RS-232/ARINC 429 CONFIG page.
2) With GIA1 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC1) and RS232 Channel 6 (GRS1) indicators are green.
3) With GIA1 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC1) and ARINC 429 IN6 (GRS1) indicators are green.
4) With GIA2 selected in the SELECT UNIT window, verify the RS232 Channel 1 (GDC2) and RS232 Channel 6 (GRS2) indicators are green.
5) With GIA2 selected in the SELECT UNIT window, verify the ARINC 429 IN5 (GDC2) and ARINC 429 IN6 (GRS2) indicators are green.
6) In the PFD page group, go to the GIA RS-232/ARINC 429 CONFIG page.
7) With PFD1 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS1) and ARINC 429 IN2 (GDC1) indicators are green.
8) With PFD2 selected in the SELECT UNIT window, verify the ARINC 429 IN1 (GRS2) and ARINC 429 IN2 (GDC2) indicators are green.

(b) PFD and MFD Ethernet connection check

1) Re-start PFD2, PFD1 and MFD in normal mode.
2) On the right hand circuit breaker panel, open the MFD circuit breaker:
3) Verify NAV1 and COM1 remain valid and NAV2 and COM2 are each replaced by a red X on both PFD1 and PFD2.
4) Close the MFD circuit breaker and wait for MFD to initialize:
5) On the right hand circuit breaker panel, open the PFD2 circuit breaker:
6) Press the DISPLAY BACK UP button above the PFD1 display.
7) Verify NAV2, COM2, NAV1, and COM1 remain valid on both the MFD and PFD1.
8) Press the DISPLAY BACK UP button above the PFD1 display to return to normal mode.
9) Close the PFD2 circuit breaker and wait for PFD2 to initialize:
10) On the left aft circuit breaker panel, open the PFD1 circuit breaker:
11) Press the DISPLAY BACK UP button above the PFD2 display.
12) Verify NAV1 and COM1 are each replaced by a red X and NAV2 and COM2 remain valid on both the MFD and PFD2.
13) Press the DISPLAY BACK UP button above the PFD2 display to return to normal mode.
14) Close the PFD1 circuit breaker and wait for PFD1 to initialize.

(c) Engine data availability to the displays and GPS data availability to the AHRS with either GIA inoperative check.

Perform "GIA Failure Test" on page 342502216.

(6) G1000 NXi Miscompare Checks

This procedure will check the AHRS, airspeed and altitude miscompare monitors. Accomplish the following checks with the aircraft positioned where it can receive GPS signals and magnetic heading.

Access to AHRS1 and AHRS2 will be required during this test.

(a) Ensure the G1000 NXi is operating in normal mode.

(b) Connect a pitot/static test set to the aircraft ADC1 pitot and static ports (Pilot's side). (Do not connect the pitot/static tester to ADC2 ports at this time)

(c) Set the baro-correction on PFD1 and PFD2 to 29.92.

(d) On the AFCS mode controller, press the AP button and verify autopilot engages.

(e) Slowly increase the pitot/static test set to simulate an ADC1 / ADC2 altitude miscompare of greater than 200 ft., verify ADC2 altitude does not increase, autopilot does not disconnect and amber comparator window text "ALT MISCOMP" is displayed on PFD1 and PFD2.

(f) Reduce ADC1 altitude to ambient pressure.

(g) Repeat Step (e) with the pitot/static test set connected to the aircraft ADC2 pitot and static ports.

(h) Use the pitot/static test set to simulate an airspeed of 40 kts for ADC1 and 55 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(i) Use the pitot/static test set to simulate an airspeed of 55 kts for ADC1 and 40 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(j) Use the pitot/static test set to simulate an airspeed of 85 kts for ADC1 and 95 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(k) Use the pitot/static test set to simulate an airspeed of 95 kts for ADC1 and 85 kts for ADC2. Verify the autopilot does not disconnect and amber comparator window text "IAS MISCOMP" is displayed on PFD1 and PFD2.

(l) Reduce ADC1 and ADC2 airspeeds to 0 kts.

(m) Slowly rotate AHRS1 along the lateral (pitch) axis to a pitch attitude of greater than five (5) degrees, verify the following:

1) AHRS2 pitch attitude does not change.

2) An amber "PIT MISCOMP" is annunciated on PFD1 and PFD2.

3) Autopilot disconnects at approximately 5 deg AHRS1 / AHRS2 miscompare.

4) The autopilot disconnect audio alert (two hi-low tones) sounds.

5) A red flashing AP annunciator on PFD1 and PFD2.

6) Flight director command bars remain in view with autopilot in HDG and PIT mode.

(n) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(o) Use the AFCS mode controller to re-engage autopilot.

(p) Slowly rotate AHRS1 along the longitudinal (roll) axis to a roll attitude of greater than five (5) degrees, verify the following:

1) AHRS2 roll attitude does not change.
2) An amber “ROLL MISCOMP” is annunciated on PFD1 and PFD2.
3) Autopilot disconnects at approximately 5 deg AHRS1 / AHRS2 miscompare.
4) The autopilot disconnect audio alert (two hi-low tones) sounds.
5) A red flashing AP annunciator on PFD1 and PFD2.
6) Flight director command bars remain in view with autopilot in HDG and PIT mode.

(q) Replace AHRS1 to normal attitude and verify that attitude display on PFD1 displays current aircraft attitude.

(r) Repeat Steps (m) through (q) for AHRS2.

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O. LRU Test Procedures

(1) PFD/MFD Test

(a) Allow displays to initialize for ~1 minute.

(b) Check that all COM/NAV display fields are valid in the top corners of both PFDs.

(c) Check that attitude, heading, altitude, airspeed, vertical speed and OAT fields are valid within 2 minutes of power up on both PFDs.

(d) Press the SENSOR softkey on each PFD and switch between ADC1 and ADC2. Verify that data from both GDC 72s is valid on both displays.

(e) Press the SENSOR softkey on each PFD and switch between AHRS1 and AHRS2. Verify that data from both GRS 79s is valid on both displays.

(f) Check that the engine instrument fields are valid on the MFD.

(g) Push the red DISPLAY BACKUP button above the pilot-side PFD. Verify that the pilot-side PFD and MFD displays enter reversion mode. MFD should have valid altitude, airspeed, vertical speed, COM1, COM2, NAV1, NAV2 and engine instruments.

(h) De-activate pilot-side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD1 and MFD return to normal display modes.

(i) Repeat Step (a) using the co-pilot-side PFD. Ensure that PFD2 and MFD enter reversion mode and MFD displays valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(j) De-activate co-pilot's side reversion mode by pushing the DISPLAY BACKUP button. Verify PFD2 and MFD return to normal display modes.

(k) If TAWS was activated, select the TAWS page (4th page in the MAP group) on the MFD.

1) Verify that the title at the top of the page reads "MAP – TAWS". If TAWS has not been enabled, the title will read "MAP – TERRAIN PROXIMITY" or "MAP – TERRAIN".

2) Press the MENU button and select "Test TAWS" from the pop-up menu.

3) After the TAWS test has completed, verify that "TAWS System Test Okay" is heard over the cockpit speaker.

(l) If no other service is to be performed, perform final return-to-service test.

(2) GMA 350c Test

(a) External power applied to aircraft. Apply power to the avionics bus.

(b) Pilot and Co-Pilot ICS. (Verifies pilot and co-pilot mic and phone jack operation.)

(c) Hold SPKR key on the Audio Panel to activate PA mode. Indicator above SPKR key will flash while PA mode is active.

(d) On pilot yoke, press Mic Key and verify audio is heard over the cockpit speaker. Press SPKR key to deactivate PA mode.

(e) Verify operation of COM 1.

Conduct VSWR check per "Chart 37" on page 342502182.

(f) Failsafe operation check for Audio Panel to #1 COM:

1) Pull AUDIO MKR circuit breaker.

2) Verify COM 1 transmits and receives normally over pilot headset.

3) Close the AUDIO MKR circuit breaker.

(g) Verify operation of COM 2. Conduct VSWR check per "Chart 37".

CHART 37
VSWR TEST

VOLTAGE STANDING WAVE RATIO (VSWR)

The standing wave ratio is a measure of the amount of power transmitted to the antenna, compared to the amount reflected back to the transmitter. S.W.R. s are a function of the transmission line, the antenna and their installation.

A. Required Equipment

Thruline Wattmeter – Model 43 bird Electronics, Cleveland, Ohio.

B. S.W.R. Testing:

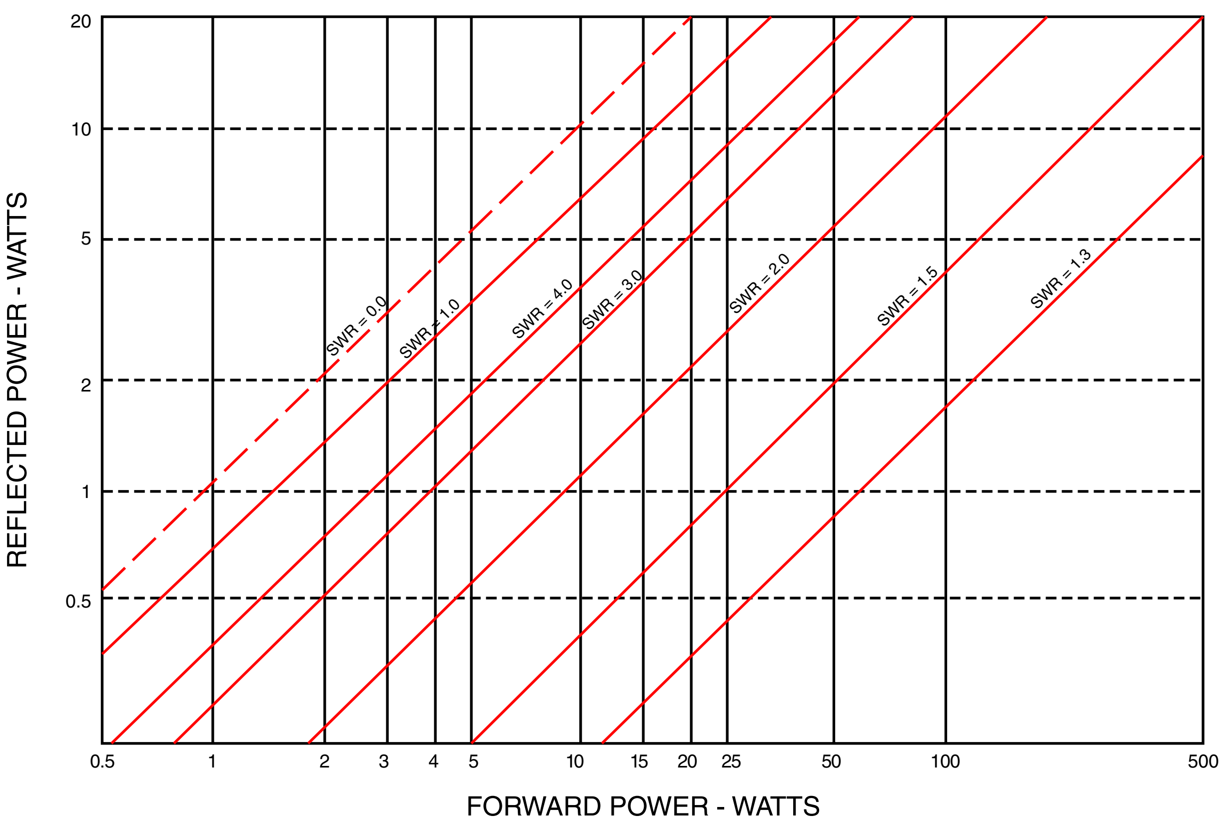
Test the antenna installation using the Thruline Wattmeter.

(1) Check VSWRs at 118.0, 123.0, 128.0, 132.0, and 135.95 MHZ by transmitting on COM 1 and then COM 2.

(2) Determine the VSWR value with the VSWR Conversion Chart below.

(3) VSWRs in excess of 2.5:1 are not acceptable.

VSWR CONVERSION CHART



(h) Marker Beacon

See "Figure 45"

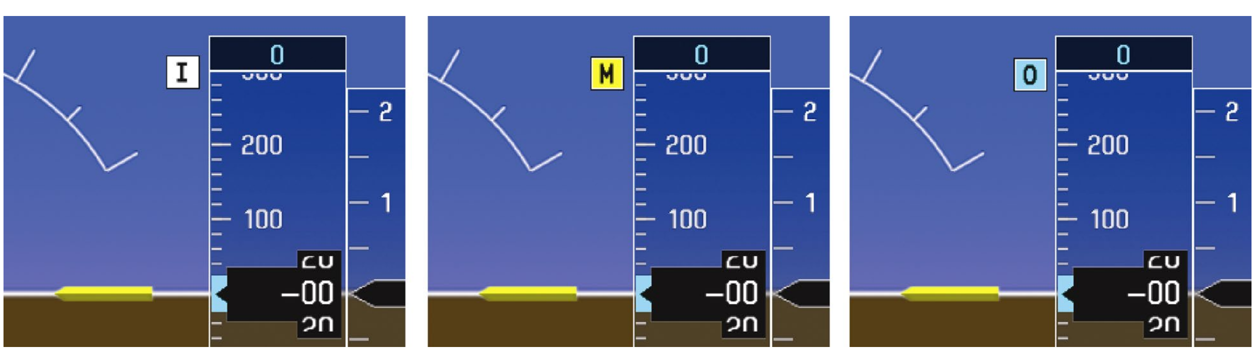
Using a ramp tester, simulate the outer marker, middle marker and inner marker signals by following the test equipment manufacturer's instructions.

1) Verify outer, middle and inner markers appear on the PFD.
2) Verify Marker Beacon Audio and tones.

   Verify that each marker audio signal is present over the pilot and co-pilot headphones and speaker.
3) Verify the MKR MUTE key mutes the Marker Beacon Audio.

(i) Verify Passenger ICS is working at all stations.

(j) If no other service is to be performed, continue to the return-to-service checks.



Marker Beacon Symbology
Figure 45

(3) GIA 64(W) Test

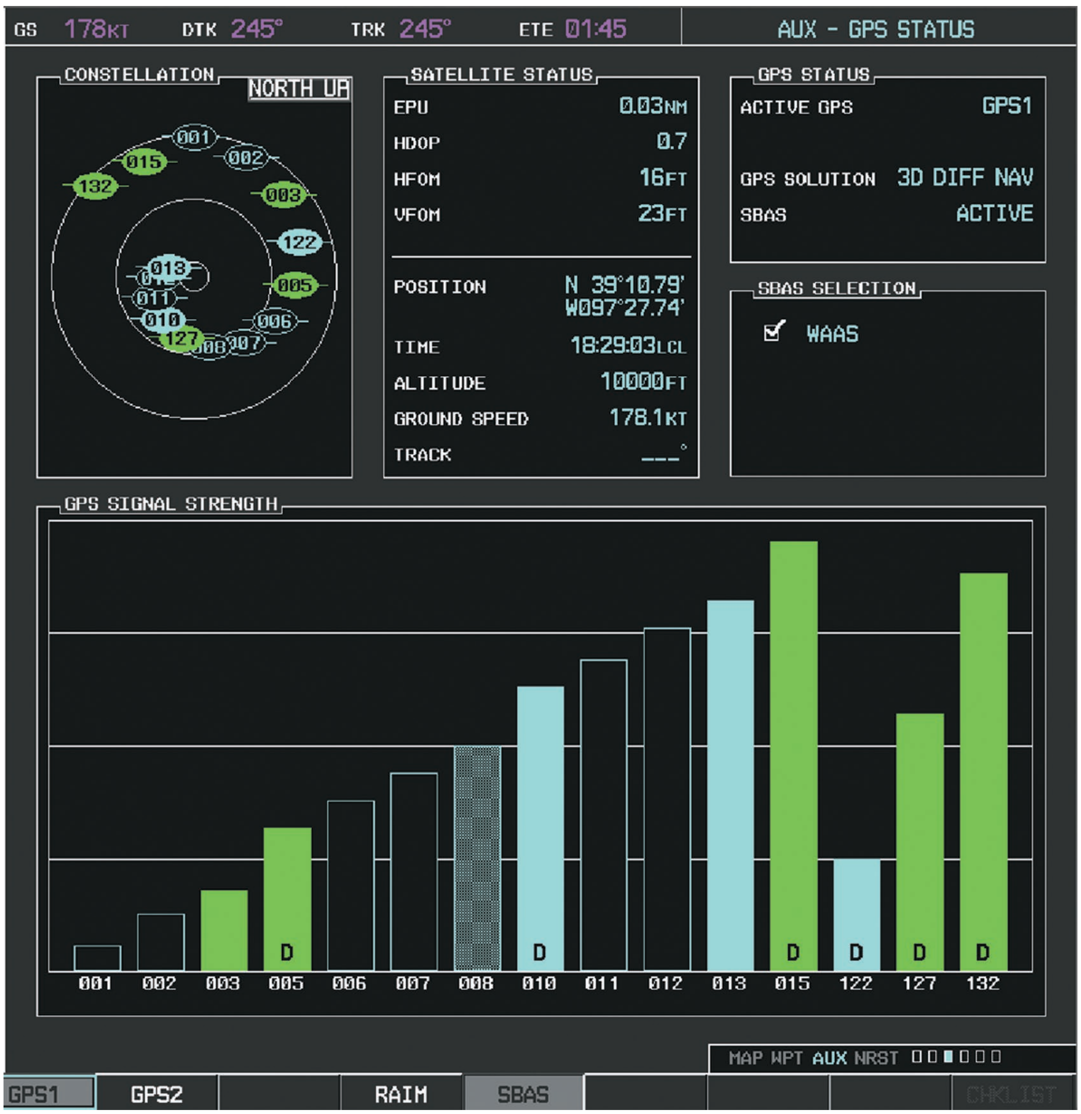
(a) GPS Signal Acquisition

See "Figure 46".

The GIA 64(W) units should normally acquire a GPS navigation solution within 2 minutes of startup, provided the aircraft is outside (or indoors with a GPS repeater). Select the GPS STATUS page on the MFD (4th page in AUX group). Two softkeys on the bottom of the display allow the user to toggle between GPS1 and GPS2. Verify that both receivers show 3D DIFF NAV on the MFD.

Continue to the VHF COM Interference test.

NOTE: It may be necessary to temporarily disable or move away from GPS repeaters while testing, as repeaters may adversely affect GPS receiver performance.



GPS STATUS Page (MFD)
Figure 46

(b) VHF COM Interference Test

This test must be conducted outside. Use of a GPS repeater inside a hangar may result in a failed test.

This procedure assumes that the system is currently set to 25 kHz COM channel spacing. Once the signal acquisition test has been completed successfully, perform the following steps:

1) On the MFD, monitor GPS signal strength bars on the AUX – GPS STATUS page.
2) On the PFD, ensure that the CDI is set to GPS. If it is not, press the ‘CDI’ softkey until GPS ENR is displayed.
3) Verify that the GPS “INTEG” flag is out of view.
4) Select 121.150 MHz on the COM1 transceiver.
5) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
6) During the transmit period, verify that the GPS “INTEG” flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
7) Repeat steps 5 and 6 and re-transmit while monitoring GPS2 signal levels on the MFD.
8) Repeat steps 4 through 7 for each of the following frequencies:
    - 121.175 MHz
    - 121.200 MHz
    - 131.250 MHz
    - 131.275 MHz
    - 131.300 MHz
9) Repeat steps 4) through 8) for the COM2 transceiver (GIA2).
10) On the MFD, select the AUX – SYSTEM SETUP page.
11) Under the COM CONFIG field, change the COM channel spacing from 25 kHz to 8.33 kHz.
12) Go back to the AUX – GPS STATUS page.
13) Select 121.185 MHz on the COM1 transceiver.
14) Transmit for a period of 35 seconds while monitoring GPS1 signal strength levels.
15) During the transmit period, verify that the GPS “INTEG” flag does not come into view on the PFD and verify that GPS1 does not lose a 3-D navigation solution on the MFD.
16) Repeat steps 14) and 15) and re-transmit while monitoring GPS2 signal levels on the MFD.
17) Repeat steps 14) through 16) for each of the following frequencies:
    - 121.190 MHz
    - 130.285 MHz
    - 131.290 Mhz
18) Repeat steps 14) through 17) for the COM2 transceiver (GIA2).
19) On the MFD, select the AUX – SYSTEM SETUP page and change the COM channel spacing back to 25 kHz.
20) Continue to the VOR/LOC/GS Test.

(c) VOR/LOC/GS Test

NOTE: The PFD HSI does not show a course deviation bar unless a valid VHF NAV frequency is tuned.

Check VOR, ILS, and Glideslope functions with ramp test equipment. Operate the equipment according to the test equipment manufacturer's instructions. Adjust the RF signal to a level adequate to perform the test. Select the appropriate HSI source by using the CDI softkey.

1) Perform NAV 1 and NAV 2 accuracy tests.
   a) Full scale.
   b) Half scale.
   c) OBS operation.
2) Check NAV 1 and NAV 2 audio and Ident.
3) Check NAV 1 and NAV 2 Sensitivity.
4) Perform LOC 1 and LOC 2 accuracy tests.
   a) Full scale.
   b) Half scale.
5) Check LOC 1 and LOC 2 Sensitivity.
6) Perform GS 1 and GS 2 accuracy tests.
   a) Full scale.
   b) Half scale.
7) Check GS 1 and GS 2 Sensitivity.

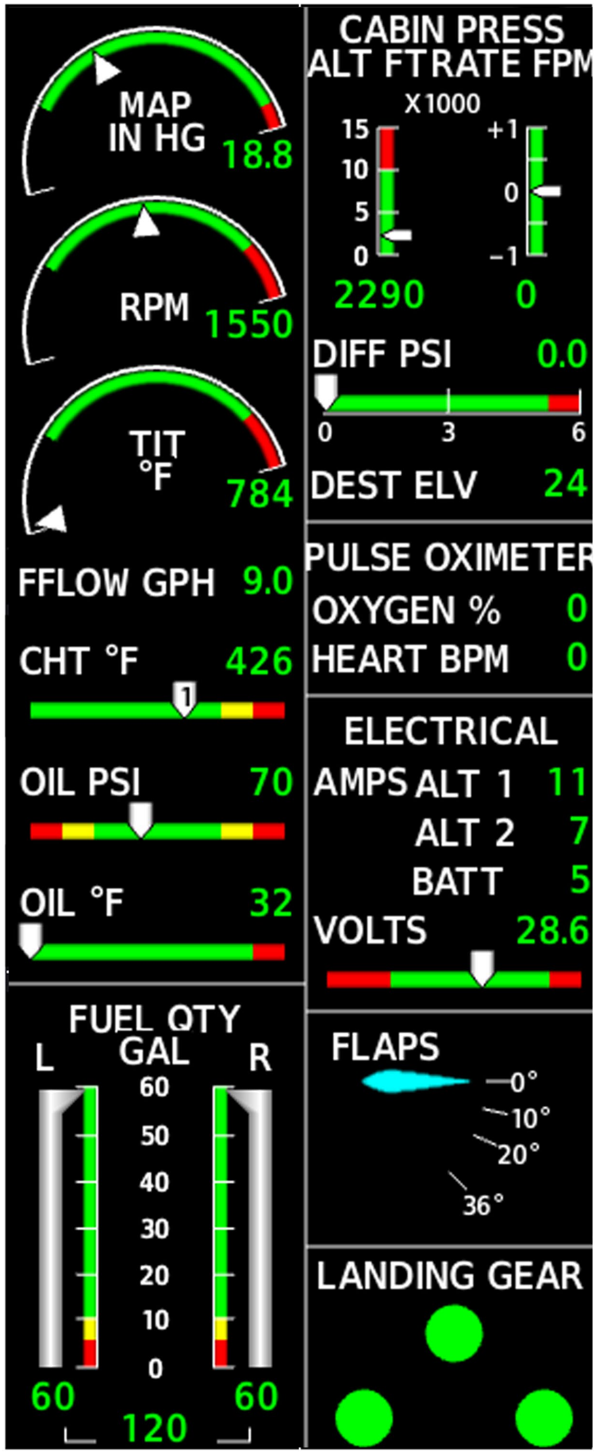
(4) GFC 700 Test

See Ground Checks under Garmin GFC 700 in 22-10-00.

(5) GEA 71B Test

(a) On the MFD (normal mode), check the indication for each of the sensor or monitor inputs with the aircraft engines off. In general, verify all engine and system instruments show valid static normal values and markings, with no red Xs or erratic indications. See "Figure 47" on page 342502187 (baseline configuration instruments shown) for normal engine instrument markings.

(b) If no other service is to be performed, continue to the return-to-service checks.



PA-46-350P

Normal Engine Instruments
Figure 47

(6) GDC 72 Air Data Computer Tests

NOTE: Allow the unit to warm up for 15 minutes before performing the following tests.

Verification of the altimeter and airspeed must be performed using an air data test set (ADTS). The static port and altimeter must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) § 91.411 and Part 43 Appendix E. The PFD must be in Configuration mode and the MFD must be in Reversionary mode for performing the tests as outlined in Part 43 Appendix E.

(a) Part 43 Appendix E Testing

1) Preparation

CAUTION: CONFIGURATION MODE CONTAINS CERTAIN PAGES AND SETTINGS THAT ARE CRITICAL TO AIRCRAFT OPERATION AND SAFETY. THESE PAGES ARE PROTECTED AND CANNOT BE MODIFIED, UNLESS THE TECHNICIAN IS PROPERLY AUTHORIZED AND EQUIPPED. HOWEVER, MOST PROTECTED PAGES ARE VIEWABLE TO ALLOW SYSTEM AWARENESS FOR TROUBLESHOOTING.

a) Start the G1000 NXi system in normal mode.

b) Remove power to PFD1.

c) Turn PFD1 on in Configuration mode by pressing and holding the ENT key on the PFD while applying power.

d) Release the ENT key after “INITIALIZING SYSTEM” appears in the upper left corner of the PFD.

NOTE: Configuration mode contains certain pages and settings that are critical to aircraft operation and safety. These pages are protected and cannot be modified, unless the technician is properly authorized and equipped. However, most protected pages are viewable to allow system awareness for troubleshooting.

2) Testing

a) Using the FMS knob on the PFD turn to the GRS page group. Use the B ALT field for all CFR Part 43 Appendix E tests for G1000 NXi altitude.

b) Place the MFD in Reversionary mode by pressing the DISPLAY BACKUP button above the respective PFD. The baro setting can then be read from the MFD for CFR Part 43 Appendix E tests.

NOTE: The baro setting on the MFD is controlled by PFD1. The baro setting will apply to both GDC1 and GDC2 regardless of the GDC selected on the MFD. The copilot’s display can be ignored.

(b) After completing the tests specified by § 91.411 and Part 43 Appendix E, return both the MFD and the PFD to normal mode.

NOTE: The following tests are above and beyond the requirements set forth in Appendix E, and are required only when appendix E tests are required.

(c) Pitot/Static Airspeed Test

1) Command air data test set (ADTS) to simulate air speeds shown in the table below.

Calibrated Air Speed (Knots)	Allowed tolerance (± Knots)
50	5.0
80	3.5
100	2.0
120	2.0
150	2.0

2) Wait for ADTS to report that target values have been achieved.

3) Verify that computed air speeds shown on the PFD are within the tolerances specified in the OEM maintenance documentation.

(d) Static Port Vertical Speed (Rate of Climb) Test

1) Command ADTS to change the altitude at the rates shown in the table below.

2) Wait for ADTS to report that target rates have been achieved.

3) Verify that the Rate of Climb reported by the Vertical Speed field on the PFD is within the tolerances specified in the table below:

Vertical Speed (feet/minute)	Allowed tolerance, (± feet/minute)
2000	100
0	45
-2000	100

(7) OAT Probe Check

Ensure on-side sensors for PFD1 and PFD2. Ensure the outside air temperature (OAT) probes and a calibrated thermometer stabilize at ambient temperature. Verify that the OAT measurement shown on PFD1 and PFD2, in degrees Celsius, indicate within 2°C of the ambient temperature as measured by the calibrated thermometer.

If no other service is to be performed, continue to the return-to-service checks.

(8) GRS 79 / GMU 44(B) Calibration Procedures

See "Chart 38" to determine when various calibrations are required. The following calibration procedures are provided for the GRS 79 AHRS and GMU 44(B) Magnetometer:

- "GRS 79 Pitch/Roll Offset Calibration (Procedure A)"
- "GRS 79/GMU 44(B) Magnetometer Calibration (Procedure B)" on page 342502193
- "Engine Run-Up AHRS Vibration Test (Procedure C)" on page 342502195
- "Magnetometer Interference Test" on page 342502196

When ready to perform the procedures, shut the PFDs and MFD off by pulling the PFD and MFD circuit breakers. Restart all displays in configuration mode. Follow the steps given for each procedure on-screen at the GRS/GMU CALIBRATION page. Note that the CALIBRATE command cannot be selected and activated until the installer acknowledges all required steps have been carried out by pressing the ENT key on each step.

CHART 38
G1000 NXi - GRS 79 AHRS/GMU 44(B) MAGNETOMETER CALIBRATIONS

Required GRS/GMU Calibrations			
	Calibrations Required		
Condition	Procedure A: GRS 79 Pitch/Roll Offset	Procedure B: GRS/GMU Magnetic Calibration	Procedure C: Engine Run-up Vibration Test
Either GMU 44(B) was removed and reinstalled. (no change in serial number)	None Required. Continue to GRS/GMU Test section.		
GMU 44(B) was replaced with new unit. (New serial number)		X	
GRS 79 AHRS was removed and/or replaced. The mounting tray was NOT removed and the mounting tray bolts were NOT loosened.	None Required. Continue to GRS/GMU Test section.		
GRS 79 AHRS was removed and/or replaced. The mounting tray WAS removed and/or mounting tray bolts WERE loosened.	X	X	X
GRS 79 AHRS Configuration Module was replaced.	X	X	X

(9) GRS 79 Pitch/Roll Offset Calibration (Procedure A)

This procedure must be performed for both GRS 79 units installed in the aircraft. This first procedure must be carried out with the engine OFF.

(a) Level the aircraft to within ±0.25° of zero pitch and zero roll using a digital level. (Refer to 8-20-00.)

(b) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(c) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(d) Select the GRS/GMU Calibration page by rotating the small FMS knob.

(e) Press the following softkeys in sequence on each PFD access the Pitch/Roll Calibration:

- 9
- 10
- 11
- 12 (far right softkey)

(f) Initiate the AHRS Ground Pitch/Roll Aircraft Level compensation mode by performing the following steps:

1) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 79 #1 ON PFD1 AND GRS 79 #2 ON PFD2.

2) Highlight the SELECT PROCEDURE window and select **PITCH/ROLL OFFSET** on each PFD.

3) Press the ENT key.

4) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

5) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

6) After several seconds, a new checklist appears in the lower half of the PFD. Press the ENT key as each step is confirmed. When the CONFIRM AIRCRAFT IS LEVEL field is blinking on both PFDs, press the ENT key to continue.

(g) The result of the pitch/roll offset compensation is displayed on each PFD. If successful, the PFDs report the successful calibration and return to normal operation.

(h) Press the ENT key on each PFD to conclude this procedure.

(i) Restart all displays in normal mode.

NOTE: The Magnetometer Calibration procedure that follows, must be carried out at a site that is determined to be free of magnetic disturbances. If it is unsure whether the site is 'clean', the technician should verify that the site is 'clean' by following the guidance provided. The technician may skip the Compass Rose Evaluation Procedure if the site condition is acceptable.

(10) Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure (Optional)

NOTE: Typically, a compass rose is an acceptable location to perform the magnetometer calibration procedure. However, because not all compass roses are well maintained, even an existing compass rose should be regularly evaluated using the method described here to determine if it is free of magnetic disturbances. If evaluation of an existing compass rose indicates that magnetic disturbances are present, then an alternative location must be found to perform the Magnetometer Calibration procedure.

A G1000 NXi-equipped airplane that has completed the pitch/roll offset compensation procedure can be used to evaluate a candidate site for magnetic disturbances and determine whether it is a suitable location to perform the magnetometer calibration procedure. The magnetometer calibration procedure itself contains the logic to simultaneously survey the location for magnetic cleanliness while it is computing the magnetometer calibration parameters. In order to evaluate a candidate site, the Magnetometer Calibration procedure must be performed twice: once turning clockwise around the site, and once turning counter-clockwise. Both times, the procedure should be conducted as described in "GRS 79/GMU 44(B) Magnetometer Calibration (Procedure B)" on page 342502193, below, with the exception of the direction of turns around the site.

NOTE: Although "GRS 79/GMU 44(B) Magnetometer Calibration (Procedure B)" on page 342502193, indicates that the Magnetometer Calibration procedure should be performed by making a series of clockwise turns around the site, the procedure can also be performed by making counterclockwise turns for the purpose of evaluating the site for magnetic disturbances.

If, upon completion of the Magnetometer Calibration procedure in both clockwise and counter-clockwise directions, the PFD displays the "CALIBRATION SUCCESSFUL / SITE IS CLEAN" message, then the candidate site is sufficiently free of magnetic disturbances and is acceptable for performing the Magnetometer Calibration procedure. It is important to obtain successful result in both the clockwise and counter-clockwise directions to ensure that the magnetometer sweeps over a large enough area at the candidate site.

If, upon completion of the Magnetometer Calibration procedure in either of the two directions, the PFD displays either the "MAG FIELD AT SITE NOT UNIFORM", or "MAG FIELD AT SITE DIFFERS FROM IGRF MODEL" message, then the site contains magnetic disturbances that are too large.

(11) GRS 79/GMU 44(B) Magnetometer Calibration (Procedure B)

NOTE: Procedure A must first be successfully accomplished before performing Procedure B, only for situations where the GRS 79 was replaced with a new unit.

Calibration Procedure B must be carried out on a compass rose in order to guarantee measurements free of environmental magnetic disturbances. Attempting to carry out this maneuver on a typical ramp area may not yield a successful calibration. The accuracy of the AHRS cannot be guaranteed if this calibration is not performed on a magnetically clean compass rose. If the compass rose condition is not known, it is recommended that the technician follow the guidance in "Compass Rose Evaluation of Magnetic Disturbances for Magnetometer Calibration Procedure (Optional)".

(a) Start the aircraft engine following the procedures in the appropriate POH.

(b) After aircraft engine startup, taxi the aircraft to a properly calibrated compass rose.

(c) At the compass rose, align the aircraft to a heading of magnetic north (±5°).

CAUTION: CALIBRATION PROCEDURE B MUST BE CARRIED OUT ON A COMPASS ROSE IN ORDER TO GUARANTEE MEASUREMENTS FREE OF ENVIRONMENTAL MAGNETIC DISTURBANCES. ATTEMPTING TO CARRY OUT THIS MANEUVER ON A TYPICAL RAMP AREA MAY NOT YIELD A SUCCESSFUL CALIBRATION. THE ACCURACY OF THE AHRS CANNOT BE GUARANTEED IF THIS CALIBRATION IS NOT PERFORMED ON A MAGNETICALLY CLEAN COMPASS ROSE OR EQUIVALENT.

NOTE: This procedure provides instructions for calibrating both GRS 79 magnetometers simultaneously by putting both PFD1 and PFD2 in configuration mode and following the procedure below, using PFD1 to calibrate GRS 79 #1 and PFD2 to calibrate GRS 79 #2.

(d) Start (restart) the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(e) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(f) Select the GRS/GMU Calibration page by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

- 9
- 10
- 11
- 12 (far right softkey)

(g) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS 79 #1 on PFD1 and GRS 79 #2 on PFD2.

(h) Highlight the SELECT PROCEDURE window and select MAGNETOMETER on each PFD.

(i) Press the ENT key.

(j) Use the cursor to highlight the BEFORE CALIBRATION window on each PFD.

(k) Follow the checklist items displayed on each PFD and press the ENT key as each step is completed or confirmed.

(l) When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(m) The PFD displays advise the operator when to turn the aircraft, when to stop, and when to turn again.

(n) Upon instruction to turn, taxi the aircraft in a right turn. After approximately 25° to 30° of turn from the last heading, the PFD displays advise the operator to stop the aircraft.

NOTE: Due to the difficulties in executing smooth, accurate turns, the PFDs may incorrectly interpret a station and instruct to "HOLD POSITION" prior to full completion of a 30° turn. If this scenario is encountered, it is best for the operator to ignore the "HOLD POSITION" command and instead use outside references to complete the approximate 30° turn. Instead of using the PFD instructions to turn as a real-time indication of when to turn, simply judge the 30° (±5°) turn increments of the aircraft by using the compass rose radials. Dwelling at these 30° increments for the time recommended by the PFDs should result in successful calibration.

(o) The PFDs guide the operator to dwell at multiple headings around a complete circle.

NOTE: Due to high winds or excessive airframe vibration, the operator may encounter a condition where a PFD restarts the 18-second countdown without full completion of the previous countdown. If this is encountered more than once for a given station, the operator should begin turning to the next station (approximately 30°). A minimum of 2 successful stations per quadrant (and for each GRS 79) is required, where a successful station is a full 18-second countdown followed by instruction to move. Ensure that at least 2 stations per quadrant are completed. Thus, it may sometimes be required to dwell at a station after a countdown restart. A maximum of 20 stations is allowed for the entire calibration procedure. If too many countdown restarts are encountered, the calibration will fail with the message, "TOO MANY STATIONS."

(p) Repeat the turn-and-stop process until each PFD advises that a successful calibration is complete. The GRS 79 AHRS units then enter the normal operational mode. Press the ENT key on each PFD to conclude this procedure.

(q) Repeat steps (d) through (q) for GRS #2, if not performed concurrently.

(12) Engine Run-Up AHRS Vibration Test (Procedure C)

NOTE: Calibration Procedure C is performed in order to guarantee that the AHRS mounting is sufficiently rigid and insensitive to vibration. This procedure must be performed for both GRS units installed in the aircraft. Calibration Procedures A and B are not required prior to this procedure.

(a) Start the PFDs and MFD in Configuration mode by holding the ENT key on each display while closing the circuit breaker.

(b) Go to the GRS Page Group on PFD #1. Press the following softkeys in sequence on each PFD:

- 9
- 10
- 11
- 12 (far right softkey)

(c) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS #1 on PFD1 and GRS #2 on PFD2.

(d) Highlight the SELECT PROCEDURE window and select ENGINE RUN-UP TEST on each PFD. Press ENT.

(e) Press the ENT key again on each PFD to select the first checklist item displayed.

(f) Follow the checklist items displayed on each PFD, and press the ENT key as each one is completed or confirmed. When the CALIBRATE field is blinking on both PFDs, press the ENT key on each PFD to begin the procedure.

(g) The PFD displays instruct the operator to gradually increase power from idle to full throttle and back to idle over a period of 1–2 minutes.

(h) When the engine run-up is completed and the engine is back to an idle setting, press the ENT key on each PFD to indicate that the process is complete. When this is done, the TEST COMPLETE field stops blinking.

(i) Each PFD will state if the installation has passed or failed the vibration test. If the test fails, the specific measurements causing the failure are identified and associated numeric values are displayed on the PFD corresponding to each GRS unit.

NOTE: Should a failure occur, the technician may perform the Engine Run-up test up to 3 times successively before corrective action must be taken. If the test does not pass after three attempts, then the installation should not be considered reliable until the source of the vibration problem is identified and remedied. In the event of repeated failure of the engine run-up test, record the values that are reported to be out of range for future reference.

The following are potential causes for failure of the engine run-up test:

1) Vibration motion of GRS and/or GMU caused by neighboring equipment and/or supports.
2) Mounting screws and other hardware for GRS and/or GMU not firmly attached.
3) GRS 79 connector not firmly attached to unit.
4) Cabling leading to GRS or GMU not firmly secured to supporting structure.
5) An engine / propeller that is significantly out of balance.

(j) Press the ENT key on each PFD to conclude this procedure.

(13) Magnetometer Interference Test

See "Figure 48" on page 342502197.

This test is used to validate that no electrical or electronic device interferes with the operation of the GMU 44 magnetometer. This test should be performed after installation or maintenance of electrical/electronic components on the aircraft and/or for troubleshooting the GMU 44.

NOTE: External power cart and other aircraft need to be as far away from the aircraft as possible.

(a) Turn Battery Master ON, AV BUS Master ON, and all circuit breakers in. Verify the Air Conditioning Condenser Blower Motor is OFF. Verify the rudder and rudder trim tab are in the streamlined position.

(b) Start the PFDs and MFD in the Configuration Mode by holding ENT on each display while closing the circuit breaker

(c) Go to the GRS Page Group on both PFDs by rotating the large FMS knob.

(d) Select the GRS/GMU Calibration page, as shown in "Figure 48", by rotating the small FMS knob. This page is protected and requires a keystroke password to perform this test. Press the following softkeys in sequence on each PFD:

- 9
- 10
- 11
- 12 (far right softkey)

(e) Activate the cursor by pressing the small FMS knob and highlight the SELECT GRS UNIT window. Select GRS #1 on PFD1 and GRS #2 on PFD2.

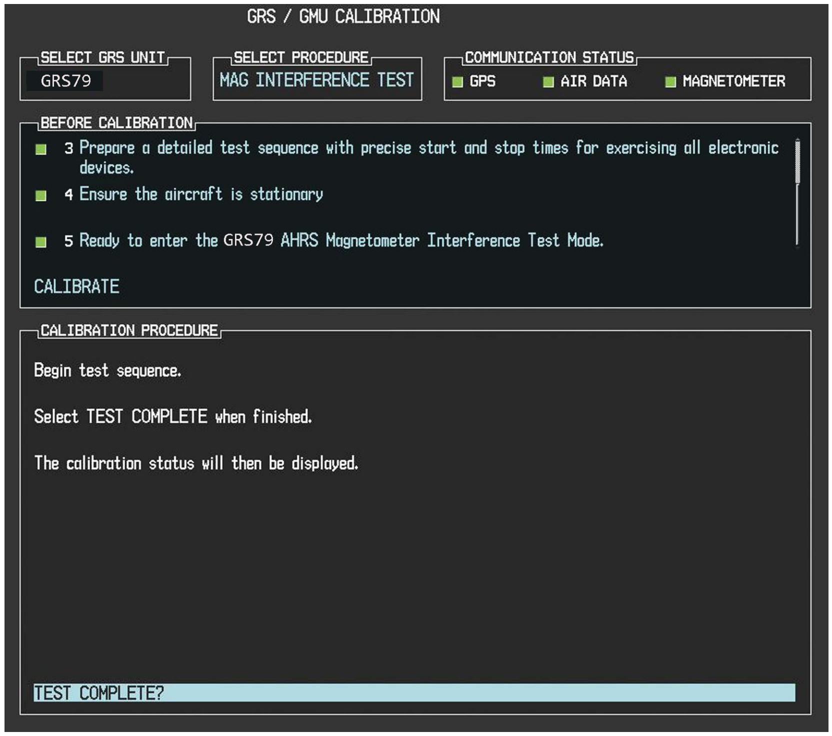
(f) Highlight the SELECT PROCEDURE window and select MAGNETOMETER INTERFERENCE TEST on each PFD.

(g) Press the ENT key to select the first checklist item displayed on each PFD. Press the ENT key as each one is completed or confirmed (turns green). Have a stopwatch ready to begin recording the elapsed time.

NOTE: The list of relevant electronic devices are given in "Chart 37". Use a stopwatch to accurately time the tests. All actions are to be carried out in the order and at the precise elapsed time as specified in the prepared test sequence.

(h) When the CALIBRATE field is blinking on both PFDs, press the ENT key to begin the procedure.

(i) When the test is complete, press the ENT key on each PFD to indicate the process is complete. When this is done, the TEST COMPLETE annunciation stops blinking.



Magnetometer Interference Test
Figure 48

CHART 39
G1000 NXi - MAGNETOMETER INTERFERENCE TEST

MAGNETOMETER INTERFERENCE TEST	
ELAPSED TIME Since start of test (Minutes:Seconds)	**ACTION**
0:00	Test Begins
0:10	Turn on strobe light
0:20	Turn off strobe light
0:30	Turn on fin strobe
0:40	Turn off fin strobe
0:50	Turn on nav lights (if optional tail light installed)
1:00	Turn off nav lights (if optional tail light installed)
1:10	Blower Motor on Low
1:20	Blower Motor off
1:30	Blower Motor on High
1:40	Blower Motor off

(j) Each PFD will display if the corresponding GRS installation has passed or failed the magnetometer interference test.

1) If the test fails, the installation should be considered unreliable until the source of magnetic interference is identified and remedied. When the magnetometer interference test fails, record the three magnetometer maximum deviation values and their corresponding timestamps. Any maximum deviation value greater than 2.5 milliGauss indicates a problem that must be resolved. Compare the corresponding timestamps with the prepared test sequence to identify which action produced the problem.

2) Two common reasons for a failed magnetometer interference test are: 1) new equipment is installed in close proximity to the GMU 44 magnetometer, and 2) an existing or new electronic device has become grounded through the aircraft structure instead of via the proper ground wire in a twisted shielded pair.

3) After troubleshooting and repeating the test, if the installation continues to fail the test contact Piper Customer Support.

(k) Press the ENT key on each PFD to conclude the test.

(14) GRS/GMU Test

The aircraft can now be taxied back and the engine can be shut down for final testing. Restart the displays in normal mode to conduct final system checks. When the PFDs power up in normal mode, the AHRS attitude and heading information displayed should become valid within 1 minute of power-up, as shown in "Figure 49" (provided both GPS receivers have a valid position; if GPS is unavailable, AHRS initialization may take as long as 2 minutes).

If no other service is to be performed, continue to the return-to-service checks.

Normal Mode AHRS Check
Figure 49

(15) GDL 69eA / 69A SXM Test

If installed and the XM Satellite Radio subscription is activated and current, proceed as follows:

(a) Power up the G1000 NXi.

(b) On the GMA, depress the small knob on the right side of the unit. LEDs on the GMA should turn blue.

(c) Press MUS1 and verify blue LEDs are on above PIL & COPIL. LED above PASS shall be off. If any LED is in the wrong state, press the applicable button to change status.

(d) Press MUS2 and verify blue LEDs are off above PIL & COPIL. Verify LED is on above PASS. If any LED is in the wrong state, press the applicable button to change status.

(e) On the MFD select AUX - XM RADIO using FMS knobs.

(f) Increment and Decrement the XM channels using the MFD controls.

(g) Increase and Decrease the volume using the MFD controls.

(h) Change the XM channel using a Bluetooth® device with Garmin Pilot connected to the FS510 (if installed). Increase and Decrease the volume using the Bluetooth® device.

(i) Verify XM audio operation at Pilot, Co-Pilot, and all passenger stations. Verify stereo operation.

(j) Plug an external audio source into the AUX jack in the cabin and verify audio can be heard in all passenger stations. Verify this audio is not present at the Pilot and Co-pilot phone jacks.

(k) If no other service is to be performed, continue to the return-to-service checks.

(16) GCU 47X Test

Perform the following key and knob presses and knob rotations on the GCU 47X, and verify the actions on the MFD.

(a) Rotate the large FMS knob and verify that the page groups change.

(b) Rotate the small FMS knob and verify that the pages change within the page groups.

(c) Use the large FMS knob to display the MAP page group and the small FMS knob to display the NAVIGATION MAP page.

(d) Rotate the RANGE knob to the right and verify the map display zooms out.

(e) Rotate the RANGE knob to the left and verify the map display zooms in.

(f) Press the RANGE knob to get the pointer on the map display.

(g) Move the RANGE knob to the left and verify the pointer moves to the left.

(h) Move the pointer up, right and down and verify that the pointer moves accordingly.

(i) Press the RANGE knob to stop displaying the pointer.

(j) Press the left and right arrowheads of the SOFTKEY SELECT keys. Verify softkeys highlighting on the MFD changes.

(k) Press the SEL key to select one of the softkeys.

(l) Press the key to display the DIRECT TO page.

(m) Use the keypad to type KIXD and verify KIXD is displayed on the MFD.

(n) Press the SPC key to add a space and then the BACK key to delete the space.

(o) Press the CLR key to clear the field.

(p) Type K34 and then press the ENTER key twice. Verify that the flight path to K34 is displayed on the map.

(q) Press FPL key to open the ACTIVE FLIGHT PLAN page. Press FPL key again to close it.

(r) Press PROC key to open the PROCEDURES page. Press PROC key again to close it.

(s) Press the MENU key to open the MENU page. Press the MENU key again to close it.

(t) If no other service is to be performed, continue to the return-to-service checks.

(17) GMC 710 Test

Perform a basic operational check on the GMC 710. The following knob, wheel and key presses and rotations are to be performed on the GMC 710.

(a) Ensure the G1000 NXi is operating in normal mode and the autopilot is operational.

(b) Press the FD key a few times; verify the Flight Director display on PFD1 toggles on and off. Leave the flight director displayed.

(c) Press the XFR key and verify the white-illuminated arrowhead points in the opposite direction.

(d) Verify the green arrow displayed at the top of PFD1 also points in the same direction.

(e) Rotate the ALT SEL knob and verify the altitude bug, displayed on PFD1 altitude tape, moves.

(f) Rotate the UP/DN wheel and verify the flight director moves in the vertical direction.

(g) Press the HDG key and verify the white illumination appears next to the key. Rotate the HDG knob and verify the heading bug, displayed on PFD1 compass card, moves and the flight director follows.

(h) Press the HDG knob and verify the heading bug centers.

(i) Press the YD key and verify the white illumination appears next to the key.

(j) Press the VS key and verify the white illumination appears next to the key.

(k) Press the FLC key and verify the white illumination appears next to the key.

(l) If no other service is to be performed, continue to the return-to-service checks.

(18) GWX 68 / 75 Test

WARNING: BEFORE ENERGIZING THE EQUIPMENT, BE SURE MICROWAVE RADIATION SAFETY PRECAUTIONS INCLUDING BOTH FUEL AND PERSONNEL SAFETY CONSIDERATIONS HAVE BEEN OBSERVED. THESE INCLUDE CLEARING ALL PERSONNEL TO AN AREA BEYOND THE MAXIMUM PERMISSIBLE EXPOSURE LEVEL (MPEL) BOUNDARY. THE MPEL FOR THE GWX 68 IS 11 FEET AND THE GWX 75 IS 12 FEET.

Operation of the GWX 68 / 75 Weather Radar is accomplished using the GCU 47X.

(a) On GWX 68:

1) Start the G1000 NXi in normal mode.

2) On the MFD, turn the large FMS knob to select the Map Page Group on the MFD, and then turn the small FMS knob to select the Weather Radar page.

3) Press the MFD FMS knob to activate the cursor in the TILT field then turn the small FMS knob to select the desired antenna tilt angle. Press the MFD FMS knob to remove the cursor.

4) Select the MODE softkey, then select the WEATHER softkey to energize the radar array. A CAUTION window will come up. After complying with the CAUTION, press YES to continue. Press the BACK softkey to show weather radar scan and gain control softkeys.

5) Select the GAIN softkey to activate the cursor in the ‘GAIN’ field. Turn the small FMS knob to adjust the gain to a non-calibrated level then press ENTER on the MFD. Verify the gain setting is visible in the gain field as a movable horizontal bar in a flashing box with a line pointer reference depicting the calibrated position. Press the FMS knob to remove the cursor.

6) Select the VERTICAL softkey, repeat paragraph 5).

7) De-select the GAIN softkey. Verify 'CALIBRATED' is displayed in the 'GAIN' field.
8) Select Horizon, Mode, and Ground softkeys sequentially. Verify scanning of and depiction of surrounding ground features. Verify Tilt, Bearing, Sector Scan, and Gain can be adjusted from the GROUND scan mode display via the FMS knob and RANGE knob, as appropriate.
9) Press the Back softkey. Select the Features softkey, then toggle the STAB softkey. Verify that STAB ON is shown in the upper right corner of the weather radar display when the STAB softkey is toggled on and STAB OFF is shown when the STAB softkey is togged off.
10) Press the Back softkey, then Mode softkey, then Standby softkey to turn off the GWX.
11) Select the OFF softkey at the conclusion of testing to turn off the weather radar.
12) If no other service is to be performed, continue to the return-to-service checks.

(b) On GWX 75:

1) Start the G1000 NXi in normal mode.
2) On the MFD, turn the large FMS knob to select the MAP Group on the MFD then turn the small FMS knob to select the WEATHER RADAR page.
3) Press the MFD FMS knob to activate the cursor in the TILT field then turn the small FMS knob to select the desired antenna tilt angle. Press the MFD FMS knob to remove the cursor.
4) Select the MODE softkey, then select the WEATHER Softkey to energize the radar array. A caution window will come up. ONLY if you have followed the precautionary steps, select yes to accept.
5) Press the BACK softkey, then press the CAL GAIN softkey to deselect "Calibrated" Gain if active.
6) Verify Tilt, Bearing, Sector Scan, and Gain can be adjusted via the FMS knob and RANGE knob as appropriate.
7) Select MODE and MAP softkeys. Verify scanning of and depiction of surrounding ground features. Verify Tilt, Bearing, Sector Scan, and Gain can be adjusted from the MAP scan mode display via the FMS knob and RANGE knob as appropriate.
8) Select MODE softkey and then WEATHER softkey.
9) Press the BACK softkey.
10) If optional GCS & TD installed, press the GCS softkey. Verify that "GND CLTR Suppress" message is displayed on the top right-hand corner of the MFD.
11) Select CAL GAIN softkey and verify that Gain Setting changes to "Calibrated".
12) Press the MODE softkey, then Standby softkey to turn off the GWX.
13) Select the OFF softkey at the conclusion of testing to turn off the weather radar.
14) If no other service is to be performed, continue to the return-to-service checks.

(19) GAE 43 Calibration

(PIR-PPS55027, Rev. New.)

The GAE 43 altitude encoder is used to generate cabin altitude and rate information for the G1000 NXi system.

(a) Required Equipment

Barfield Pitot Static Test Set, P/N 101-00164, (or equivalent) with male 1/8-27 NPT fitting plumbed to end of static line.

(b) Requirements

Each 24 months and whenever the GAE 43 encoder unit is replaced, calibrate the new unit before installation per procedure, below.

(c) Procedure

CAUTION: USE A BACKUP WRENCH ON THE FEMALE 1/8-27 NPT PIPE FITTING ON THE REAR OF THE GAE-43 WHEN ATTACHING THE STATIC LINE FROM THE TEST UNIT TO AVOID DAMAGING THE GAE-43 UNIT.

1) Wrap the male end of the test set static line in Teflon tape (MMM No. 547 or equivalent) and connect to the female 1/8-27 NPT pipe fitting on the rear of the GAE 43 (see "Figure 20" on page 34250297).

2) Apply ground power and start the MFD and PFDs in normal mode.

3) Set the test set pressure altitude to 25,000 ft.

4) Adjust the GAE 43 such that the cabin pressure altitude indicated on the MFD is within 20 ft of the altitude set on the tester, as follows:

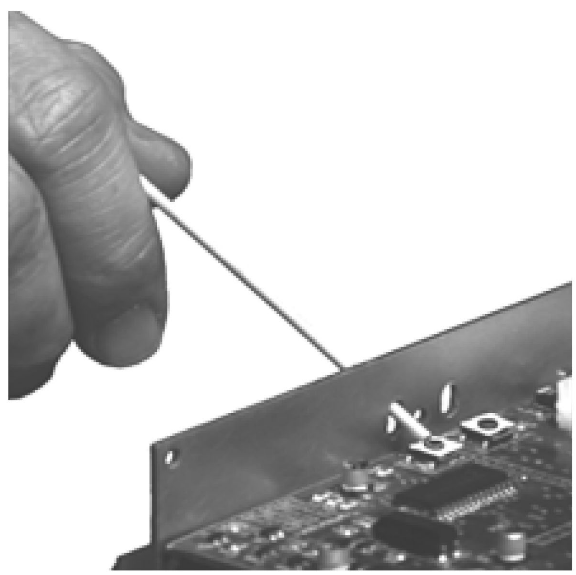
a) To increase cabin pressure altitude indicated on the MFD:

Press the increase button behind slot A on the rear of the GAE 43, using a non-conductive tool ("Figure 50").

b) To decrease cabin pressure altitude indicated on the MFD:

Press the decrease button behind slot B on the rear of the GAE 43, using a non-conductive tool ("Figure 50").

5) Set the test set pressure altitude to -900 ft. If, due to ambient atmospheric conditions, the test set is not capable of reaching -900 ft altitude, use the lowest negative altitude achievable on the test set as a calibration point.



Adjusting GAE 43
Figure 50

NOTE: The MFD altitude indicated in step 6 should be calibrated to ± 20 ft of the actual achievable altitude applied via the test set, if it differs from -900 ft.

6) Adjust the GAE 43 such that the cabin pressure altitude indicated on the MFD is within 20 ft of the altitude set on the tester, per step 4).

7) Set the test set pressure altitude to 5,000 ft.

8) Adjust the GAE-43 such that the cabin pressure altitude indicated on the MFD is within 20 ft of the altitude set on the tester, per step 4).

9) Repeat steps 7) and 8) at 10,000 ft and 12,500 ft test set pressure altitudes.

10) Remove power from the G1000 NXi system. Disconnect the test set static line from the rear of the GAE 43 unit.

(20) Stormscope (WX-500) Functional Check

If installed.

(a) Start the G1000 NXi system in normal mode.

(b) On the MFD, turn the large FMS knob to select the MAP group, then turn the small FMS knob to select the STORMSCOPE page.

(c) During initialization, the WX-500 stormscope runs a series of self-tests to ensure that all functions are operating properly.

1) If no faults are detected continue to step 4.

2) If a fault is detected, consult the WX-500 Installation manual for troubleshooting assistance.

(d) Verify that the range can be changed from 25 NM to 200 NM via the range knob.

(21) TAWS Functional Check

If installed:

(a) Basic

3) Select the TAWS-B page (last page in the MAP group) in the MFD.

4) Verify that the title at the top of the page reads “MAP - “TAWS-B”. If TAWS has not been enabled, the title will read “MAP - TERRAIN PROXIMITY” or “MAP - TERRAIN-SVT”.

5) Press the MENU button and select “Test TAWS SYSTEM” from the pop-up menu and press ENT.

6) After the TAWS test has completed, verify that “TAWS System Test Okay” is annunciated over the audio system.

(b) Detailed

1) With the G1000 NXi in Normal Mode, use the GCU FMS knob to select the MAP group and TAWS page on the MFD.

2) Verify that the title at the top of the page reads “MAP – TAWS”.

NOTE: If TAWS has not been enabled, the title will read “MAP – TERRAIN PROXIMITY” or “MAP – TERRAIN”. Refer to TAWS Configuration for configuring TAWS.

3) Press the GCU MENU button and select “Test TAWS” from the pop-up menu. Verify TAWS test annunciation is displayed on the MFD and both PFDs.

4) After the TAWS test has completed, verify that “TAWS System Test Okay” is heard over the cockpit speaker.

5) Press the GCU MENU button again and select “Inhibit TAWS” from the pop-up menu and press ENT on the GCU. Verify “TAWS INHB” is displayed on PFD1 and PFD2.

6) Press the GCU MENU button again and select “Enable TAWS” from the prop-up menu and press ENT on the GCU. Verify the “TAWS INHB” annunciation on the PFDs has extinguished.

7) With a GPS position acquired, shield or disconnect the GPS antennas to remove the GPS signal. Verify “DR” shows on the MFD and the “TAWS N/A” and “DR” annunciations show on the PFDs.

8) Reconnect or remove the shield from the GPS antennas, and verify the MFD "DR" indication and PFD "TAWS N/A" and "DR" annunciations are removed once the GPS satellites are acquired.
9) Pull PFD1 PRI and PFD1 SEC circuit breakers. Re-power PFD1 in configuration mode, and use the PFD1 FMS knob to select the Audio Alert Configuration page.
10) Ensure cockpit speaker is selected ON. Use the PFD1 FMS knob to highlight each of the following messages then select PLAY. Verify each of the following audio messages can be played:
    - PDA – Caution: Too Low Terrain
    - EDR – Caution: Sink Rate
    - EDR – Warning: Pull Up
    - NCR – Caution: Don't Sink
    - VCO – Caution: Five Hundred
    - RTC – Caution: Caution, Terrain (2X)
    - RTC – Warning: Terrain (2X), Pull Up (2X)
    - ROC – Caution: Caution, Obstacle (2X)
    - ROC – Warning: Obstacle (2X), Pull Up (2X)
    - ITI – Caution: Terrain Ahead (2X)
    - ITI – Warning: Terrain Ahead, Pull Up (2X)
    - IOI – Caution: Obstacle Ahead (2X)
    - IOI – Warning: Obstacle Ahead, Pull Up (2X)
11) Pull the PFD1 PRI and PFD1 SEC circuit breakers, and re-power PFD1 in normal operating mode.
12) If no other service is to be performed, continue to the return-to-service checks

(22) Terrain Proximity Check

If TAWS is not installed.

(a) Ensure that matching terrain data base cards are installed in the bottom card slots in the PFDs and MFD and the aircraft has a GPS position.

(b) On the MFD, select the MAP group, TERRAIN PROXIMITY page by rotating the large FMS knob to change groups and the small FMS knob to change pages.

(c) Verify the title at the top of the page reads "MAP – TERRAIN PROXIMITY".

(d) Verify that the map is colored to show terrain and no failures are annunciated on the page.

(23) SurfaceWatch Check

If Installed.

(a) Navigate to the System Setup page of the Aux group on the MFD.

(b) Activate the cursor and use the large FMS knob to scroll to SurfaceWatch Alerts and set it to "Off" and verify the system message "SURFACEWATCH INHIBITED".

(c) Set SurfaceWatch Alerts to "On" and verify "SURFACEWATCH INHIBITED" system message clears.

(24) ChartView Functional Check

If installed.

ChartView must be enabled, see "CHARTVIEW Unlock (Optional)" on page 342502163 for enabling procedures, if required. A current database must be installed as well. See "JEPPS NAV Database and Charts Procedure" on page 342502166, if required.

(a) Basic

1) Select the Airport Information page (first page in the WPT group) in the MFD.

2) Verify that the title at the top of the page reads “WPT- Airport Information”.

3) Enter KVRB as the waypoint and verify that the CHRT softkey is available.

4) Press the CHRT softkey and verify the electronic chart is viewable.

(b) Detailed

1) With the G1000 NXi in Normal Mode, use the GCU FMS knob to select ‘AUX – System Status’ page and verify ChartView database cycle number is displayed in blue text and ChartView database is current.

2) From the Navigation Map page, press the SHW CHRT softkey and verify airport chart is displayed and the following softkeys are displayed:

- CHRT OPT
- CHRT
- INFO
- DP
- STAR
- APR
- WX
- NOTAM
- GO BACK

3) Press CHRT OPT softkey and verify softkeys advance to the following level of softkeys:

- ALL
- HEADER
- PLAN
- PROFILE
- MINIMUMS
- FIT WDTH
- FULL SCN
- BACK

4) Return to the Navigation Map Page.

5) Press the GCU MENU key to display the PAGE MENU. Turn the GCU large FMS knob to scroll through the OPTIONS Menu to ‘Show Chart’. Press the GCU ENT key to display the chart and verify airport diagram is being displayed.

6) Press the GCU FMS knob to activate the cursor. Turn the GCU large FMS Knob to select the Airport Identifier Box.

7) Turn the GCU small and large FMS knob to enter the airport identifier for New Century airport (KIXD) then press the GCU ENT key to complete the airport selection.

8) Select the APR softkey. Turn the GCU large FMS Knob to select the Approach Box then turn the small FMS Knob to show the approach chart selection choices.

9) Turn either GCU FMS knob to scroll through the available charts and select a chart for viewing by pressing the GCU ENT key to complete the chart selection and verify the appropriate ChartView chart is displayed.

10) If no other service is to be performed, continue to the return-to-service checks.

(25) Synthetic Vision Check

If installed.

NOTE: You will need a valid GPS signal.

(a) Press the PFD softkey on the #1 PFD, then press the SVT softkey.

(b) Press the TERRAIN softkey.

(c) Verify the traditional blue/brown attitude depiction is replaced with the Synthetic Vision rendering within 2–3 minutes.

(26) SafeTaxi Functional Check

If installed.

The maximum map ranges for enhanced detail are configurable by the flight crew. When zoomed in close enough to show the airport detail, the map reveals runways with numbers, taxiways with identifying letters/numbers, and airport landmarks including ramps, buildings, control towers, and other prominent features. Resolution is greater at lower map ranges. When the aircraft location is within the screen boundary, including within SafeTaxi ranges, an airplane symbol is shown on any of the navigation map views for enhanced position awareness. Any map page that displays the navigation view can also show the SafeTaxi airport layout, within the maximum configured range. The following is a list of pages where the SafeTaxi feature can be seen:

- Navigation Map Page
- Inset map
- Weather Datalink Page
- Airport Information Page
- Intersection Information Page
- NDB Information Page
- VOR Information Page
- User Waypoint Information Page
- Trip Planning Page
- Nearest Pages

(a) Use the FMS knob on the GCU to select the AUX – System Status page and select DBASE softkey. Use the small FMS knob to scroll to CHART category and verify “ChartView” is displayed in blue text adjacent to “CHART”.

(b) Verify the Chartview database ‘REGION’, ‘CYCLE’ number, ‘EFFECTIVE’, ‘EXPIRES’, and ‘DISABLES’ dates of the subscription appear in blue text.

(c) Use the FMS knob on the GCU to select MAP - Navigation Map page.

(d) On the GCU, press MENU. With Map Setup highlighted, press ENT on the GCU. Rotate the small GCU FMS knob to select the Aviation group and press the ENT key on GCU.

(e) Turn the GCU large FMS Knob to scroll through the Aviation Group options to ‘SAFETAXI’.

(f) Turn the GCU small FMS Knob to display the range of distances.

(g) Turn the GCU FMS Knob to select 5000ft as the distance for maximum SafeTaxi display range, and then press the GCU ENT key to complete the selection.

(h) Using the GCU range knob, select a range of 5000ft or less. Verify SafeTaxi display represents the current aircraft location and the airport layout. If no other service is to be performed, continue to the return-to-service.

(27) GTS 825

If Installed.

(a) System Check

1) Start the G1000 NXi system in normal mode and select the TRAFFIC MAP page on the MFD MAP group.

2) Turn ADS-B off by pressing the ADS-B softkey if the softkey indicator is green.

3) Verify that a TAS Status (not TAS FAIL) is displayed in the upper right corner of the traffic map. Verify that a yellow NO DATA message is not displayed in the center of the map. If TIS is displayed instead of TAS, refer to "Optional Equipment Configuration" on page 342502158, to install the GTS 825 option.

4) Press the TAS OPER softkey and verify that OPERATING is displayed in the upper right corner of the traffic map.

5) Press the TAS STBY softkey and verify that STANDBY is displayed in the upper right corner of the traffic map.

6) Set range to 6NM or greater.

7) Press the TEST softkey and verify that TEST is displayed in the upper left corner of the traffic map and that a traffic test pattern is displayed. Verify that “TAS System Test” aural is heard over the cockpit speaker at the conclusion of the test.

8) Open the TAS circuit breaker and verify that FAILED is displayed at the center of the traffic map after several seconds.

9) Close the TAS circuit breaker and verity that FAILED is removed from the traffic map.

(b) Function Test

Perform the following test to verify GTS 825 operational and surveillance functions. Use a ramp tester such as a TIC TR220 or equivalent to perform the tests.

1) To select a scenario that will properly converge and intercept the GTS 825, the GTS 825 must be in ground test mode. To enable ground test mode, the aircraft must be on the ground and the GTS 825 must be in normal system mode and in standby.

2) Position the test set directional antenna with a clear line of sight to the GTS 825 antenna at 90 degrees. With the GTS 825 powered up and in standby mode indicated on the CDTI, cycle the GTS 825 to ‘Operate.’

3) Select the following:

a) Set the intruder type as ATCRBS.

b) Intruder Start Distance: 10 nm.

c) Intruder Start Altitude: 50,000 ft.

d) Vertical Speed: 0 fpm.

e) Velocity: 360 kts.

4) Initiate the intruder scenario and observe the following:
   a) Traffic should be acquired at approximately 10 NM at 90 degree bearing and co-altitude. Observe intruder closes on own aircraft at a rate of 0.1 NM/sec.
   b) The intruder should transition from Other Traffic (displayed as an open diamond with 00 displayed above), to proximate traffic (displayed as a filled white diamond with 00 displayed above), to a Traffic Advisory (TA) alarm.
   c) The appropriate TA symbology (yellow filled circle with 00 displayed above, and an audio annunciation of “Traffic! 3 O’clock! At Altitude! 3 Miles!”), displayed when the intruder approaches within 3 NM.

(28) GSR 56 Checkout

If Installed.

NOTE: The following verifies correct installation of GSR 56. See "Chart 28" on page 342502120, if the Iridium Satellite Radio/Datalink requires activation.

NOTE: The telephone button must be active on the GMA 350C to hear audio from the GSR 56.

(a) On the MFD navigate to AUX group, SATELLITE PHONE page using the FMS knobs.
(b) Press the DIAL softkey.
(c) Use the softkeys or FMS knobs to enter “1911135” and press ENT to confirm.
(d) Press ENT to call.
(e) If both parties can hear each other, then the installation has been verified. Tell the representative that you are testing the GSR 56 installation and that the unit is active.

NOTE: If placed on hold when calling Garmin Flight Data Services, you may call anyone for a brief verification phone call. Normal charges will apply.

(29) FS 510 Checkout

If installed.

NOTE: This section requires the use of a Bluetooth® enabled device, such as a smart phone or tablet with the Garmin Pilot app installed.

(a) Ensure the airplane has a valid GPS position.
(b) Turn on Bluetooth® on the mobile device.
(c) Navigate to the Aux group, Connext Setup page on the MFD. Note the Bluetooth® unique identifier at the end of the Bluetooth® name field on this page.
(d) Pair to “Flight Stream 510-XXXX” on the mobile device, where XXXX is the unique identifier for this particular Flight Stream 510.
(e) Follow prompts on the device as well as on the MFD to confirm the confirmation code.
(f) Open the Garmin Pilot application on the mobile device.
(g) Tap the button in the top left corner of the mobile device.
(h) Select Connext from the list on the left of the mobile device screen, then select Flight Stream 510 from the devices list.
(i) Select Flight Stream 510-XXXX from the dropdown menu on the mobile device and then press Connect.
(j) After the device has connected, press Activate on the mobile device to receive the flight plan update.
(k) Tap the button in the top left corner of the mobile device twice and then select Flight Plan.
(l) Ensure that KVRB appears as the first airport listed.
(m) Tap “Add Waypoint” and type in KFPR and press done.

(n) Tap the blue Connext icon located to the right of Flight Plan on the top row on mobile device.

(o) Tap "Send to Navigator".

(p) Press ENT on the MFD when prompted with "pending flight plan."

(q) Press ENT on the MFD when prompted to activate and press ENT again to click OK.

(r) Go to the Navigation Map on the MFD and verify flight path is correct.

(30) ADF Adjustment and Checkout

If installed.

Select an appropriate local NDB (Station 1) and determine the station's bearing from the test location.

(a) Quadrantal Error Adjustment

1) Position aircraft on ramp in an area that is clear of surrounding buildings and other obstructions that may block or reflect radio signals.

2) Select ADF as the bearing source on the #1 PFD.

3) Select ADF mode on ADF receiver.

4) Tune the ADF to Station 1.

5) Orient aircraft such that the bearing indicator points to 0 degrees (aircraft is heading directly to Station 1). Note the heading (Direction A).

6) Using the heading indicator, rotate the aircraft 45 degrees to the right (Direction A plus 45 degrees).

7) ADF indicator should show 315 ±3 degrees.

8) Using the heading indicator, rotate the aircraft 90 degrees to the left (Direction A minus 45 degrees).

9) ADF indicator should show 45 ±3 degrees.

NOTE: The following steps are not required if the quadrantal error is within tolerance.

10) Adjust the variable resistor R60 on the rear side of the RMI-Converter to correct the quadrantal error.

11) Recheck the relative bearings and readjust the QE variable resistor as necessary to split the errors at the quadrantal points and obtain the lowest possible average error.

(b) Checkout

1) Tune the ADF to Station 1 and verify the bearing is with +/- 15° of the known bearing to Station 1 and the Morse Code Identifier is clear and accurate.

2) Verify the ADF audio is available on the GMA 350c audio panel.

(c) If no other service is to be performed, continue to the return-to-service checks.

(31) DME Functional Check

If installed.

(a) Perform systems check with appropriate DME test equipment (i.e., ATC 600, etc.) and verify the DME is tuned by NAV 1 and NAV 2.

(b) Verify the distance is accurate.

(c) Verify the DME audio is available on the GMA audio panel (via AUX buttton).

(d) If no other service is to be performed, continue to the return-to-service checks.

(32) GTX 3X5R Test

Operation of the GTX 3X5R Mode S/ES transponder is accomplished using PFD1, PFD2 or the MFD. Refer to the appropriate G1000 NXi Cockpit Reference Guide, for basic operation.

The integrated transponder/altitude reporting system must be verified in accordance with Title 14 of the Code of Federal Regulations (CFR) §§ 91.411 and 91.413, every 24 calendar months, or any time the transponder is removed. This test requires the use of a Mode S ramp generator. Specific instructions for operating the ramp tester are contained in the applicable operator's manual. Refer to Title 14 CFR Part 43.

(33) Display System Checkout

(a) NAV Source Selection Tests

NOTE: The CDI will not be depicted if the VOR antenna is not receiving a valid signal.

NOTE: For FMS, the word that is displayed with the FMS (i.e., FMS Term) depends on the airplane's phase of flight. Either FMS TERM or FMS ENR is acceptable.

1) Start these tests with all avionics powered.

2) Press the Direct-To-Key (-D>).

3) Enter a Direct-To waypoint. Press ENT to accept waypoint.

4) Enter 117.3 MHz (TRV VOR), an appropriate local VOR frequency, or an appropriate VOR frequency simulated with test equipment, into the active NAV 1 frequency box on #1 PFD.

5) Enter 110.0 MHz (MLB VOR), an appropriate local VOR frequency, or an appropriate VOR frequency simulated with test equipment, into the active NAV 2 frequency box on #1 PFD.

6) On MFD, AUX Group, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight CDI in the Synchronization window. Turn the small FMS knob to turn synchronization ON. Press the FMS knob to deactivate the cursor.

7) Verify that PFD1 and PFD2 synchronize (i.e., PFD1 & PFD2 match for the CDI).

8) Press the CDI softkey on PFD1. Verify the NAV source on PFD1 and PFD2 changes from FMS TERM to VOR1.

9) Press the CDI softkey on PFD1. Verify the NAV source on PFD1 and PFD2 changes from VOR1 to VOR2.

10) Press the CDI softkey on PFD1. Verify the NAV source on PFD1 and PFD2 changes from VOR2 to FMS TERM.

11) Repeat Step 6), but turn the synchronization OFF.

12) By pressing the CDI softkey on PFD1, cycle the NAV source on PFD1 from FMS TERM to VOR1 to VOR2 TO FMS TERM. Verify that only the NAV source depicted on PFD1 changes from FMS TERM to VOR1 to VOR2 to FMS TERM.

13) By pressing the CDI softkey on PFD2, cycle the NAV source on PFD2 from FMS TERM to VOR1 to VOR2 to FMS TERM. Verify that only the NAV source depicted on PFD2 changes from FMS TERM to VOR1 to VOR2 to FMS TERM.

(b) Bearing Pointers and Information Windows Tests

NOTE: The bearing pointer will not be depicted if it is not receiving a valid signal.

1) Start these tests with all avionics powered.
2) Press the Direct-To-Key (-D->).
3) Enter a Direct-To waypoint. Press ENT to accept waypoint.
4) Enter 117.3 MHz (TRV VOR), an appropriate local VOR frequency, or an appropriate VOR frequency simulated with test equipment, into the NAV1 frequency box on PFD1.
5) Enter 110.0 MHz (MLB VOR), an appropriate local VOR frequency, or an appropriate VOR frequency simulated with test equipment, into the NAV2 frequency box on PFD1.
6) Enter 257 kHz (SQT NDB) or an appropriate local ADF frequency into the ADF receiver (if installed).
7) Press the PFD softkey on PFD1 to make the BRG1 and BRG2 softkey selections visible.
8) Verify that the BRG1 softkey can be pressed to sequentially cycle through NAV1, GPS, and ADF (if installed).
9) Verify that the BRG2 softkey can be pressed to sequentially cycle through NAV2, GPS, and ADF (if installed).
10) Repeat Steps 7)–9) for PFD2.

(c) BARO Selection Tests

1) On MFD AUX Group, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization ON. Press the FMS knob to deactivate the cursor.
2) Rotate the BARO knob on PFD1 and verify that both PFD’s BARO values change.
3) Rotate the BARO knob on PFD2 and verify that both PFD’s BARO values change.
4) Set the current altimeter setting and verify that the both PFD’s indicate within ±30 ft. of local field elevation.
5) On MFD AUX Group, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization OFF. Press the FMS knob to deactivate the cursor.
6) Rotate the BARO knob on PFD1 and verify that only PFD1 BARO values change and a delta indication of at least 0.01 in. Hg turns the indication yellow.
7) Rotate the BARO knob on PFD2 and verify that only PFD2 BARO values change and a delta indication of at least 0.01 in. Hg turns the indication yellow.
8) On MFD AUX Group, SYSTEM SETUP, press the FMS knob to activate the cursor. Turn the large FMS knob to highlight BARO in the Synchronization window. Turn the small FMS knob to turn synchronization on. Press the FMS knob to deactivate the cursor.

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P. Return to Service Procedure

WARNING: SOFTWARE VERSIONS MUST BE CHECKED AND MATCHED AGAINST THE VERSIONS LISTED IN THE APPROPRIATE REQUIRED EQUIPMENT LIST (REL), SEE "CHART 41" ON PAGE 342502227. SOFTWARE CONFIGURATION IS A CRITICAL PART OF G1000 NXI OPERATION AND MUST BE VERIFIED BEFORE RETURNING AN AIRCRAFT TO SERVICE.

(1) Software Verification

After reinstalling any G1000 NXi LRU, use the checkoff list in "Chart 40" on page 342502226 to verify correct LRU software part numbers and versions against the current numbers and versions shown in the appropriate Required Equipment List (REL) ("Chart 41" on page 342502227).

(a) Start the G1000 NXi system in configuration mode.

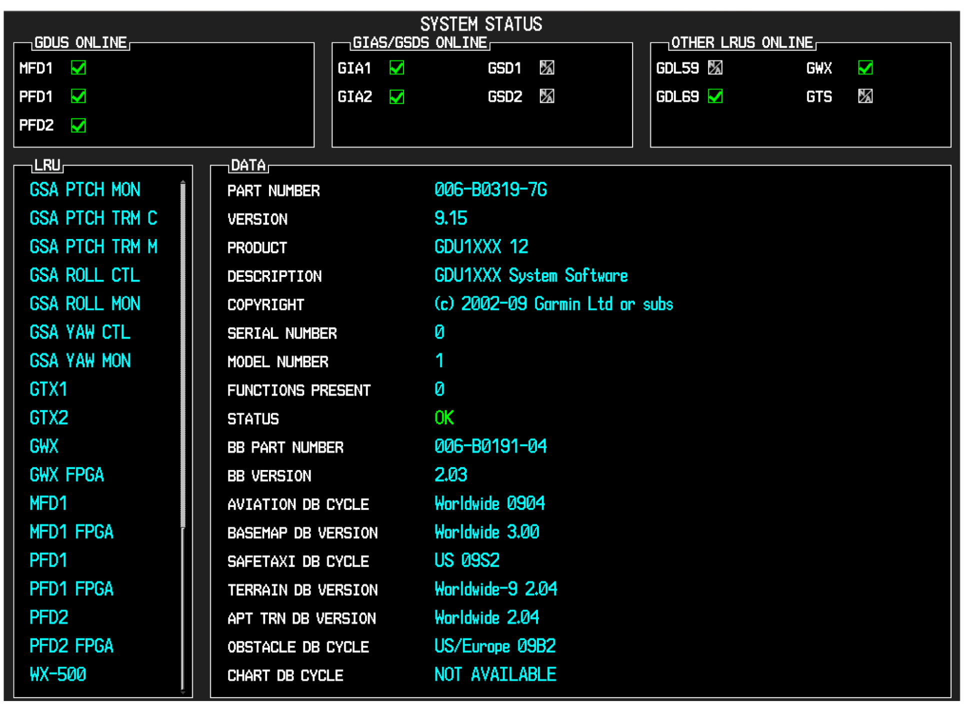
(b) The PFD System Status page (see "Figure 51") displays a list of LRUs in the LRU window.

(c) The MFD initial power up page (i.e., splash screen) displays the installed software image number (see the "System" number in the top right of "Figure 51" on page 342502213).

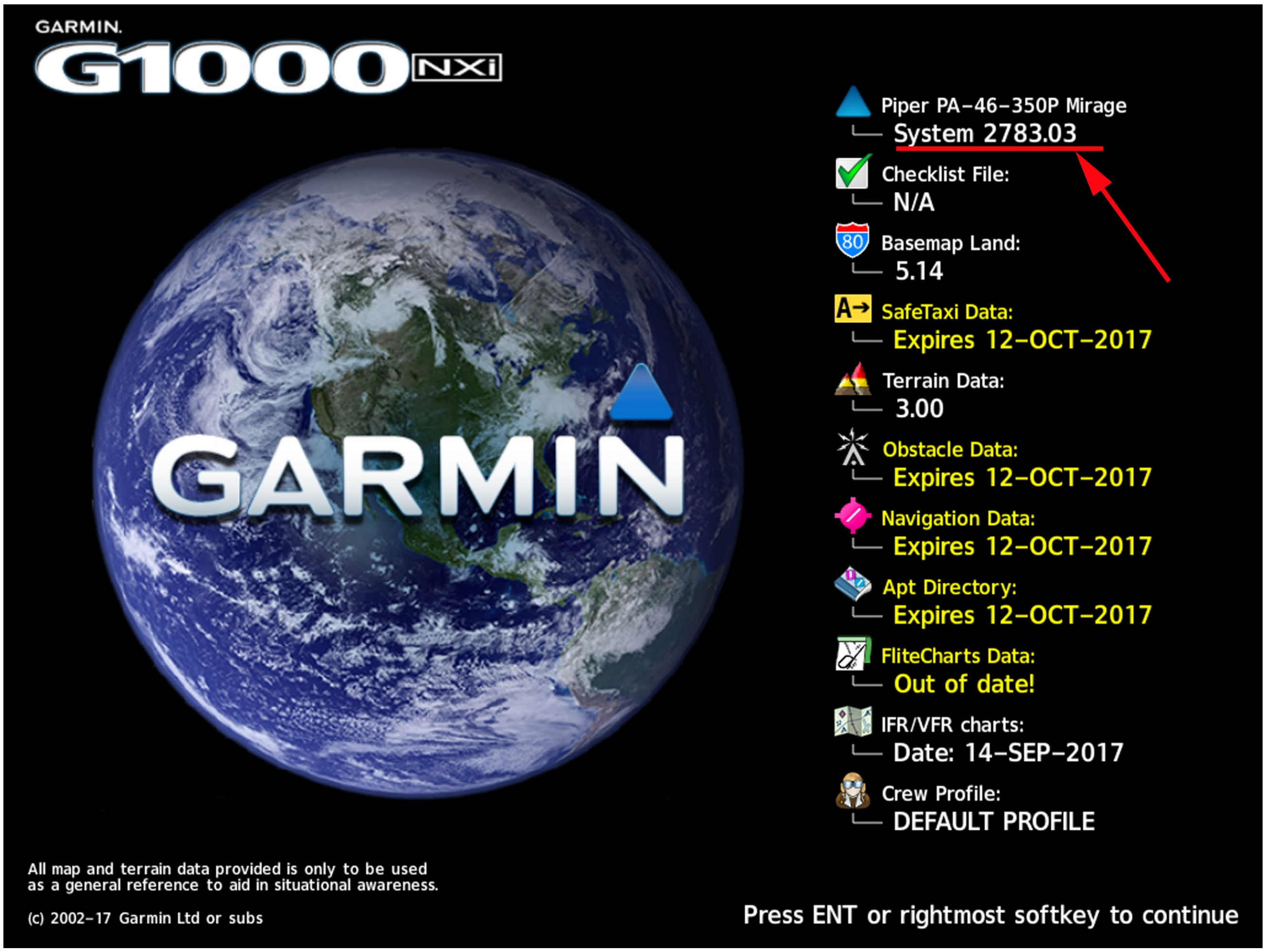
(d) Activate the cursor and highlight the LRU window.

(e) Use the FMS knob to scroll through the list, selecting each LRU in turn.

(f) The software part number and version is displayed in the DATA window. Compare this to the current numbers and versions shown in the appropriate Required Equipment List (REL) ("Chart 41" on page 342502227).



System Status Page (Configuration Mode)
Figure 51



MFD Initial Power Up Page
Figure 52

NOTE: The following Failure Tests test various secondary communications paths to ensure that the desired backup paths are in place.

(2) GPS Failure Test

(a) Start these tests with all avionics powered, Valid GPS reception. See "Figure 46" and "GPS Signal Acquisition" on page 342502184.

(b) Press Direct-To-Key (~~D~~>). Enter a Direct-To waypoint.

(c) Place a shroud over the GPS 1 antenna to prevent signal reception. Verify loss of GPS 1 signal on MFD AUX group, GPS STATUS page. The following should remain valid on the PFD throughout the procedure.

1) Attitude
2) Heading
3) Airspeed
4) Altitude
5) Vertical Speed
6) OAT
7) GPS CDI remains valid on PFD.

(d) Remove shroud from the GPS 1 antenna.

(e) Place a shroud over the GPS 2 antenna to prevent signal reception. Verify loss of GPS 2 signal on MFD AUX group, GPS STATUS page. The following should remain valid on the PFD throughout the procedure.

1) Attitude
2) Heading
3) Airspeed
4) Altitude
5) Vertical Speed
6) OAT
7) GPS CDI remains valid on PFD.

(f) Leave the shroud over the GPS 2 antenna and replace the shroud over the GPS 1 antenna. Verify loss of GPS 1 and 2 signals on MFD AUX page 4. The following should remain valid on the PFD throughout the procedure.

1) Attitude
2) Heading
3) Airspeed
4) Altitude
5) Vertical Speed
6) OAT
7) NO FMS POSITION annunciated on the PFD inset map.

(3) GIA Failure Test
   (a) Start these tests with all avionics powered, valid GPS reception, valid NAV signal.
   (b) Engage the autopilot by pressing the AP key on AFCS mode controller.
   (c) If XFR button is not pointing left, couple FD to PFD1 by pressing the XFR button on the AFCS mode controller.
   (d) Remove power from GIA 1 by pulling the INTEG AV 1 and COM 1 circuit breakers.
   (e) Verify the following:
      1) Autopilot and yaw damper disconnect with a continuous AP disconnect tone.
      2) The FD bars are removed.
      3) A red-boxed AFCS Annunciation appears on the PFD.
   (f) Acknowledge the tone by pressing the AP DISC TRIM INTER switch.
   (g) Verify the following system messages:
      1) AHRS1 GPS – AHRS1 USING BACKUP GPS SOURCE
      2) FAILED PATH
      3) AHRS2 GPS – AHRS2 NOT RECEIVING BACKUP GPS INFORMATION
      4) XPDR1 FAIL (If dual GTX 33 option is installed)
      5) GMA1 FAIL
      6) GAE 43 FAILURE
      7) ESP FAIL (If Enhanced AFCS installed)
   (h) Verify the following flag invalid:
      1) COM/NAV 1 field
      2) NAV 1 CDI loses deviation bar.
      3) XPDR 1 FAIL (If dual GTX 33 option is installed).
   (i) Verify BOTH ON GPS2 Sensor Reversion annunciation on the PFDs.
   (j) Restore power to GIA1 by resetting the INTEG AV 1 and COM 1 circuit breakers.
   (k) Verify the following:
      1) All invalid flags, system messages, and annunciations are removed. (#1 GPS will take time to reinitialize).
      2) Autopilot passes the Preflight Test (PFT).
   (l) Engage the autopilot by pressing the AP key on AFCS mode controller.
   (m) Remove power from GIA 2 by pulling the INTEG AV 2 and COM 2 circuit breakers.
   (n) Verify the following:
      1) Autopilot and yaw damper disconnect with a continuous AP disconnect tone.
      2) The FD bars remain in view.
      3) A red-boxed AFCS Annunciation appears on the PFD.
   (o) Acknowledge the tone by pressing the AP DISC TRIM INTER switch.
   (p) Verify the following system messages:
      1) AHRS2 GPS – AHRS1 USING BACKUP GPS SOURCE
      2) STRMSCP FAIL (if installed)
      3) AHRS1 GPS – AHRS1 NOT RECEIVING BACKUP GPS INFORMATION.
      4) FAILED PATH
      5) CO DET FAIL

6) ESP FAIL (If Enhanced AFCS installed)

(q) Verify the following flag invalid:

1) COM/NAV 2 field
2) NAV 2 CDI loses deviation bar.
3) Stormscope (on MFD Stormscope Map page) via STORMSCOPE FAILED yellow text (if installed).

(r) Verify BOTH ON GPS1 Sensor Reversion annunciation on the PFDs.

(s) Restore power to GIA2 by resetting the INTEG AV 2 and COM 2 circuit breakers.

(t) Verify the following:

1) All invalid flags, system messages, and annunciations are removed. (#2 GPS will take time to reinitialize).
2) Autopilot passes the Preflight Test (PFT).

(u) Remove power from GIA 1 and GIA 2 by pulling the INTEG AV 1, INTEG AV 2, COM 1 and COM 2 circuit breakers.

1) Verify the following system messages:
   a) LOSS OF GPS NAVIGATION
   b) XPDR1 FAIL
   c) GMA1 FAIL
   d) TRN AUD FAIL (If TAWS is installed)
2) Verify the following flags invalid:
   a) COM/NAV 1 field
   b) COM/NAV 2 field
   c) XPDR field
   d) Engine Instruments
   e) Cabin Alt, Diff Pressure, and Cabin Rate
3) Verify the following:
   a) All AHRS and ADC fields remain valid.
   b) NAV 1 & NAV 2 CDI lose deviation bar.
   c) “GEAR SYS” CAS warning.
   d) “CPCS FAIL” CAS caution (“CPCS FAULT” may also annunciate)

(4) Display Failure Test
   (a) Start these tests with all avionics powered.
   (b) Select PFD on the GCU 47X Keypad. Verify FD is coupled to PFD1 by a left pointing arrow on the AFCS mode controller next to the XFR button.
   (c) Engage the autopilot by pressing the AP key on the AFCS mode controller.
   (d) Remove power from the MFD by pulling the MFD circuit breaker.
   (e) Verify the following system messages on the PFDs:
      1) GDL69 FAIL (If Installed)
      2) FAILED PATH
      3) ESP OFF (if Enhanced AFCS installed)
   (f) Verify the following:
      1) COM/NAV 2 fields flag invalid.
      2) ADF indication (BRG1 & BRG2) flag invalid (if option is installed.
      3) DME indication flags invalid (if option is installed)
      4) GMC 710 Autopilot controller remains functional while XFR is to PFD #1.
      5) PFDs switch to reversion mode.
      6) Attitude and Heading remain valid from AHRS.
      7) Airspeed, Altitude, Vertical Speed and OAT remain valid.
      8) Engine Instrumentation appears on PFDs.
   (g) Verify the autopilot retains the following flight director modes:
      1) HDG
      2) ROL
      3) NAV (need a valid VOR signal)
      4) ALT
      5) VS
      6) FLC
   (h) Restore power to the MFD. The displays should return to Normal operation.
   (i) Activate co-pilot's DISPLAY BACKUP switch (reversionary mode on #1 PFD).
   (j) Select MFD on the GCU 47X Keypad.
   (k) Couple FD to PFD # 2 by pressing the XFR button on the AFCS controller.
   (l) Remove power from the #1 PFD by pulling the PFD1 circuit breaker.
   (m) Verify the following system messages are displayed on #2 PFD and the MFD:
      1) FAILED PATH
      2) TCAS FAIL (If GTS 825 installed.)
   (n) Verify the following:
      1) MFD and PFD2 switch to reversion mode.
      2) Attitude and Heading remain valid from AHRS.
      3) Airspeed, Altitude, Vertical Speed and OAT remain valid.
      4) MFD retains engine instrumentation.
      5) GCU 47X Keypad functions on MFD.
      6) GMC 710 Autopilot Controller remains functional while XFR is to #2 PFD.

(o) Restore power to the #1 PFD and deactivate the co-pilot's DISPLAY BACKUP switch. The displays should return to normal operation.

(p) Couple FD to PFD # 1 by pressing the XFR button on the AFCS controller.

(q) Select reversionary mode on #2 PFD by activating pilot's DISPLAY BACKUP switch.

(r) Remove power from the #2 PFD by pulling the PFD2 circuit breaker.

(s) Verify the following system messages are displayed on #1 PFD and the MFD:

1) FAILED PATH
2) XPDR2 ADS-B TRFC (if installed)
3) XPDR2 ADS-B NO POS (if installed)
4) XPDR2 CSA FAIL (if installed)
5) GWX FAIL (if installed)

(t) Verify the following:

1) MFD and #1 PFD switch to reversion mode.
2) Attitude and Heading remain valid from AHRS.
3) Airspeed, Altitude, Vertical Speed and OAT remain valid.
4) MFD retains engine instrumentation.
5) GMC710 Autopilot controller remains functional while XFR is to PFD #1.

(u) Restore power to the #2 PFD and deactivate pilot's DISPLAY BACKUP switch. The displays should return to normal operation.

(5) AHRS Failure Tests

NOTE: Allow for the AHRS to realign between tests.

(a) Start these tests with all avionics powered.

(b) Engage the autopilot by pressing the AP key on the AFCS mode controller

(c) Remove power from the #1 AHRS by pulling the AHRS 1 circuit breaker.

(d) Verify the autopilot and yaw damper disconnect with a continuous AP disconnect tone.

(e) Acknowledge the tone by pressing the AP DISC TRIM INTER switch.

(f) Verify the following Comparator Annunciations are displayed on the PFDs:

1) HDG NO COMP
2) ROL NO COMP
3) PIT NO COMP

(g) Verify Heading, Pitch, and Roll flag invalid on #1 PFD.

(h) Press the SENSOR key on #1 PFD.

(i) Press the AHRS2 key on #1 PFD.

(j) Verify the following:

1) #2 AHRS parameters are displayed on #1 PFD without invalid flags.
2) BOTH ON AHRS2 sensor reversion annunciation is displayed on the PFDs.
3) White NO COMP Annunciations remain on the PFDs.

(k) Restore power to the #1 AHRS.

(l) Select AHRS1 key on #1 PFD.

(m) Verify normal operation, failure and Comparator Annunciations are removed.

(n) Repeat steps (b) through (m) for the #2 AHRS and #2 PFD.

(6) Cross-Side AHRS Test
   (a) Start this test with all avionics powered.
   (b) Press the Sensor key on each PFD.
   (c) Press the AHRS2 Key on #1 PFD.
   (d) Press the AHRS1 Key on #2 PFD.
   (e) Verify the USING AHRS 2 annunciation on #1 PFD and USING AHRS 1 annunciation on #2 PFD.
   (f) Press the AHRS1 Key on #1 PFD.
   (g) Press the AHRS2 Key on #2 PFD.
   (h) Verify AHRS annunciations are removed.
(7) Air Data Computer (ADC) Failure Tests
   (a) Start these tests with all avionics powered.
   (b) Engage the autopilot by pressing the AP key on AFCS mode controller.
   (c) Remove power from the #1 ADC by pulling the #1 ADC circuit breaker.
   (d) Verify the autopilot and yaw damper remain engaged. Disconnect autopilot by pressing the AP button on the AFCS mode controller.
   (e) Verify the following Comparator Annunciations are displayed on the PFDs:
      1) IAS NO COMP
      2) ALT NO COMP
   (f) Verify the following flag invalid #1 PFD.
      1) Altitude
      2) Indicated Airspeed
      3) Vertical Speed
      4) Static Air Temperature (OAT) and ISA
   (g) Press the SENSOR key on #1 PFD.
   (h) Press the ADC2 key on #1 PFD.
   (i) Verify the following:
      1) #2 ADC parameters are displayed on #1 PFD without invalid flags.
      2) BOTH ON ADC2 sensor reversion annunciation is displayed on the PFDs.
      3) White NO COMP Annunciations remain on the PFDs.
   (j) Restore power to the #1 ADC.
   (k) Select ADC1 key on #1 PFD.
   (l) Verify normal operation, invalid flags and Comparator Annunciations are removed.
   (m) Repeat steps (b) through (l) for the #2 ADC and #2 PFD.

(8) AHRS/Air Data Backup Path Test

(a) Start these tests with all avionics powered.

(b) Remove power from the #1 PFD by pulling the #1 PFD circuit breaker.

(c) On #2 PFD, press the SENSOR soft key and select ADC1 and AHRS1. This test verifies the secondary AHRS/ADC 1 paths. The following shall occur during this test:

1) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.

2) Engine Instrumentation remains valid.

3) COM/NAV 1 fields remain valid.

(d) Restore power to the #1 PFD.

(e) Remove power from the MFD and GIA 1 by pulling the MFD and INTEG AV 1 circuit breakers.

(f) On #1 PFD, press the SENSOR softkey and select ADC1 and AHRS1. This test verifies the primary AHRS/ADC 1 paths. The following shall occur during this test:

1) Attitude, Heading, Airspeed, Altitude, Vertical Speed and OAT remain valid.

2) COM/NAV1 and COM/NAV 2 fields flag invalid.

3) All engine instruments flag invalid.

(g) Restore power to the MFD and GIA 1.

(9) Display Testing

The G1000 NXi system is tested while operating in the normal mode unless otherwise specified. If the system is in configuration mode, restart the displays by cycling the PFD1, PFD2 and MFD circuit breakers to start the display in the normal mode.

(a) Apply aircraft power. Observe the MFD power-up screen. Verify the display is formatted as shown in "Figure 52" on page 342502214. This figure is a format reference, the illustration on the left may vary. Note database versions and system software represented in the upper right corner. Refer to the appropriate Required Equipment Lists (REL), "Chart 41" on page 342502227, for the correct database and system software versions.

(b) The ‘System’ number reflected in the upper right hand corner is the Software Image Version. It correlates to G1000 NXi SW Loader Card used to load the software to the system.

(c) Verify that the System Image Version is correct per the appropriate Required Equipment List (REL), "Chart 41" on page 342502227.

(d) Press the GCU ENT key to acknowledge the correct pilot profile on the MFD.

NOTE: The rightmost softkey on the MFD may also be used.

(e) In the normal operating mode, data fields that are invalid have large red X's through them ("Figure 53"). A valid field does not display a red X ("Figure 54" on page 342502223). Allow the displays to initialize for approximately one minute. The GDC 72s requires a longer initialization period than do the other LRUs. During normal operation, this causes the airspeed, altitude, vertical speed, and OAT fields to be invalid during the first 40–60 seconds of PFD power-up.

(f) Check that all COM/NAV fields are valid in the top corners of PFD1 and PFD2.

(g) Check altitude, airspeed, vertical speed, TAS, and OAT fields are valid on PFD1 and PFD2.

(h) Press the SENSOR softkey on each PFD and switch between ADC1 and ADC2. Verify that data from both GDCs are valid on both displays.

(i) Press the SENSOR softkey on each PFD and switch between AHRS1 and AHRS2. Verify that data from both GRS 79s are valid on both displays.

(j) Check that engine instrument fields are valid on the MFD.

(k) Verify that no MANIFEST alert messages appear in the lower right corner (press the flashing ALERTS softkey to view alert messages). If any MANIFEST errors appear, the correct software to the related LRU must be loaded before proceeding.

PFD Power Up System Annunciations
Figure 53

PFD Operation
Figure 54

(10) Reversion Mode Check

(a) Push the DISPLAY BACKUP button/switch on the pilot's side. Verify that the pilot-side PFD and MFD displays enter reversion mode (see "Figure 55"). MFD should have valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(b) De-activate pilot-side reversion mode. Verify PFD1 and MFD return to normal display modes.

(c) Repeat Step 1 using the DISPLAY BACKUP button/switch on the co-pilot's side. Ensure that PFD2 and MFD enter reversion mode and MFD displays valid altitude, airspeed, vertical speed, COMM1, COMM2, NAV1, NAV2 and engine instruments.

(d) De-activate co-pilot's side reversion mode. Verify PFD2 and MFD return to normal display modes.

(e) Open PFD1 circuit breaker. Verify the PFD1 goes blank and MFD display remains in normal mode.

(f) Activate pilot-side reversion mode. Verify that the MFD displays enter reversion mode. MFD should have valid altitude, airspeed, vertical speed, COMM2, NAV2 and engine instruments.

(g) Close PFD circuit breaker and de-activate pilot-side reversion mode. Verify PFD1 and MFD return to normal display modes.

Reversionary Mode
Figure 55

(11) Engine Run Checks

The following verify certain annunciations and displays.

WARNING: DO NOT WORK IN THE ENGINE AREA WHILE THE ENGINE IS RUNNING.

CAUTION: AFTER RUNNING THE ENGINE, THE EXHAUST STACKS WILL BE HOT.

(a) Remove fuses F411 and F412. (Simulates alternator failure.)
(b) START engine.
(c) Verify Alternator Operation:
1) Turn ON Alternator. Verify ALTR 1 FAIL and ALTR 2 FAIL annunciation.
2) Turn OFF Engine
3) Reinstall fuses F411 and F412.
4) Restart engine.
5) Verify proper alternator operation.
(d) Verify MAP, RPM, TIT, Fuel Flow, CHT, Oil Pressure, and Oil Temp indications are operating in normal range.
(e) Verify alternator #1 and alternator #2 current indications.
(f) With the Battery Master Switch ON and Alternator #1 and Alternator #2 on line, turn the Battery Master and Alternator #1 OFF.
1) Verify BATTERY OFF advisory CAS message annunciates (if installed).
2) Record Alternator #2 current.
3) Turn the Alternator #1 ON and alternator # 2 off.
4) Record Alternator #1 current.
5) Compare Alternator #1 and #2 current readings and verify they are equal (within 3 amps).
6) Turn on alternator #2. Turn on battery master and verify BATTERY OFF advisory clears (if installed).

(12) Flight Test

A flight test is recommended after the installation is complete to ensure satisfactory performance of the G1000 NXi.

(13) VHF Comm Tests

Verify the communications capability on both the high and low ends of the VHF COM spectrum.

(14) VOR/ILS Tests

Select a VOR channel within a 40 nautical mile range. Listen to the VOR audio and verify that no electrical interference such as magneto noise is present. Check the tone identifier filter operation. Fly inbound or outbound on a selected VOR radial and check for proper LEFT / RIGHT, TO / FROM flag indications on the CDI. Check the VOR accuracy.

(15) Maintenance Records

After conducting the function check flight, the airplane may be returned to service. Record the following information in the appropriate airplane maintenance logs:

(a) Part number of the loader card used to perform software loading or software updates.
(b) Any other applicable information related to the maintenance work performed on the airplane.

CHART 40
G1000 NXi - RETURN TO SERVICE CHECK LIST

Group System	SW Ver. OK	System	SW Ver. OK	System	SW Ver. OK
System Software					
PA-46 Image File	______				
Control & Display:					
PFD1	______	PFD2 FPGA	______	GCU	______
PFD1 FPGA	______	MFD1	______	GMC	______
PFD2	______	MFD1 FPGA	______		
CNS:					
GIA1	______	GTX1	______	COM1	______
GIA2	______	GTX2	______	COM2	______
GMA1	______	GIA1 AUDIO	______	NAV1	______
GMA2	______	GIA2 AUDIO	______	NAV1 FPGA	______
GPS1	______	GDL69 (1)	______	NAV2	______
GPS2	______	GWX	______	NAV2 FPGA	______
GS1	______	GWX FPGA	______		
GS2	______				
Sensors:					
GDC1	______	GMU1	______	GRS1	______
GDC1 FPGA	______	GMU1 FPGA	______	GRS1 FPGA	______
GDC2	______	GMU2	______	GRS2	______
GDC2 FPGA	______	GMU2 FPGA	______	GRS2 FPGA	______
GEA1	______				
AFCS:					
GSA PTCH	______	GSA ROLL	______		
GSA PTCH TRM	______	GSA YAW	______		
AFCS Cert Gains:					
GFC CERT GIA1	______	GFC CERT PT C	______	GFC CERT YC	______
GFC CERT GIA2	______	GFC CERT PT M	______	GFC CERT YM	______
GFC CERT PC	______	GFC CERT RC	______		
GFC CERT PM	______	GFC CERT RM	______		

(1) If installed.

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.03 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B2371-02	2.02	
COM2	006-B2371-02	2.02	
GCU1	006-B0742-12	3.10	
GDC1	006-B1838-53	2.03	
GDC1 RGN LIST	006-D5080-50	2.00	
GDC2	006-B1838-53	2.03	
GDC2 RGN LIST	006-D5080-50	2.00	
GDL69	006-B1902-05	5.20	Optional, Verify If Installed
GEA1	006-B2139-02	2.10	
GIA1	006-B2548-05	2.02	
GIA1 AUDIO	006-D0425-22	4.02	
GIA2	006-B2548-05	2.02	
GIA2 AUDIO	006-D0425-22	4.02	
GMA1	006-B2103-03	4.21C	
GMA1 AUDIO CONFIG	006-D3035-B8	4.22C	
GMA1 REGION LIST	006-D3035-77	4.10C	
GMA1 AUX	006-B2104-04	4.21C	
GMA1 AUX ASR CONFIG 1	006-D3034-33	4.20	
GMA1 AUX ASR CONFIG 2	006-D3034-08	2.00	
GMA1 AUX ASR CONFIG 3	006-D3034-09	2.00	
GMA1 AUX AUDIO DATABASE	006-D3034-36	4.20C	
GMA1 AUX REGION LIST	006-D3034-29	4.10C	
GMC1	006-B0387-20	3.00	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	
GMU2 FPGA	006-C0048-00	2.00	
GPS1	006-B1827-10	7.0	
GPS2	006-B1827-10	7.0	
GRS1	006-B1838-03	2.03	
GRS1 REGION LIST	006-D5080-00	2.00	

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.03 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GRS2	006-B1838-03	2.03	
GRS2 REGION LIST	006-D5080-00	2.00	
GSA PTCH CTL	006-B0398-39	3.41	
AFCS GAINS	006-D2018-02	2.02	
GSA PTCH MON	006-B0398-39	3.41	
AFCS GAINS	006-B2018-02	2.02	
GSA PTCH TRM C	006-B0398-39	3.41	
AFCS GAINS	006-D5863-01	2.01	
GSA PTCH TRM M	006-B0398-39	3.41	
AFCS GAINS	006-D5863-01	2.01	
GSA ROLL CTL	006-B0398-39	3.41	
AFCS GAINS	006-D2018-02	2.02	
GSA ROLL MON	006-B0398-39	3.41	
AFCS GAINS	006-D2018-02	2.02	
GSA YAW CTL	006-B0398-39	3.41	
AFCS GAINS	006-D2018-02	2.02	
GSA YAW MON	006-B0398-39	3.41	
AFCS GAINS	006-D2018-02	2.02	
GTS	006-B1244-55	3.05	Optional, Verify If Installed
GTS FPGA	006-C0081-20	2.0	Optional, Verify If Installed
GTS RGN LIST	006-D2767-02	3.00	Optional, Verify If Installed
GTX1	006-B0172-14	8.04	Optional, Verify If Installed (GTX 33D)
GTX1	006-B1607-08	2.11	Optional, Verify If Installed (GTX 335)
GTX2	006-B1607-08	2.11	Optional, Verify If Installed
GWX	006-B0266-12	2.20	Optional, Verify If Installed
GWX FPGA	006-C0042-06	3.01	Optional, Verify If Installed
MFD1	006-B1177-64	20.80	
NAV1	006-B2253-00	2.00	
NAV2	006-B2253-00	2.00	

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.03 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
PFD1	006-B1177-64	20.80
PFD2	006-B1177-64	20.80
PA-46-350P IMAGE	006-B2783-03	2783.03 (1-2)

(1) PIR-107997, Rev. A, Factory installed in 4636716, 4636720–4636768.
(2) See "Figure 52" on page 342502214.

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.04 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B2371-02	2.02	
COM2	006-B2371-02	2.02	
GCU1	006-B0742-12	3.10	
GDC1	006-B1838-56	2.06	
GDC1 RGN LIST	006-D5080-50	2.00	
GDC2	006-B1838-56	2.06	
GDC2 RGN LIST	006-D5080-50	2.00	
GDL69	006-B1902-0A	5.50	Optional, Verify If Installed
GEA1	006-B2139-05	2.21	
GIA1	006-B2548-0L	2.07	
GIA1 AUDIO	006-D0425-27	4.07	
GIA2	006-B2548-0L	2.07	
GIA2 AUDIO	006-D0425-27	4.07	
GMA1	006-B2103-05	5.20C	
GMA1 AUDIO CONFIG	006-D3035-EP	5.21C	
GMA1 REGION LIST	006-D3035-D8	5.20C	
GMA1 AUX	006-B2104-05	5.20C	
GMA1 AUX ASR CONFIG 1	006-D3034-33	4.20	
GMA1 AUX ASR CONFIG 2	006-D3034-08	2.00	
GMA1 AUX ASR CONFIG 3	006-D3034-09	2.00	
GMA1 AUX AUDIO DATABASE	006-D3034-36	4.20C	
GMA1 AUX REGION LIST	006-D3034-B7	5.20C	
GMC1	006-B0387-20	3.00	
GMU1	006-B0224-01	2.05	
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	
GMU2 FPGA	006-C0048-00	2.00	
GPS1	006-B1827-10	7.0	
GPS2	006-B1827-10	7.0	
GRS1	006-B1838-07	2.07	
GRS1 REGION LIST	006-D5080-00	2.00	

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.04 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GRS2	006-B1838-07	2.07	
GRS2 REGION LIST	006-D5080-00	2.00	
GSA PTCH CTL	006-B0398-43	3.53	
AFCS GAINS	006-D2018-02	2.02	
GSA PTCH MON	006-B0398-43	3.53	
AFCS GAINS	006-B2018-02	2.02	
GSA PTCH TRM C	006-B0398-43	3.53	
AFCS GAINS	006-D5863-01	2.01	
GSA PTCH TRM M	006-B0398-43	3.53	
AFCS GAINS	006-D5863-01	2.01	
GSA ROLL CTL	006-B0398-43	3.53	
AFCS GAINS	006-D2018-02	2.02	
GSA ROLL MON	006-B0398-43	3.53	
AFCS GAINS	006-D2018-02	2.02	
GSA YAW CTL	006-B0398-43	3.53	
AFCS GAINS	006-D2018-02	2.02	
GSA YAW MON	006-B0398-43	3.53	
AFCS GAINS	006-D2018-02	2.02	
GTS	006-B1244-5A	3.14	Optional, Verify If Installed
GTS FPGA	006-C0081-20	2.0	Optional, Verify If Installed
GTS RGN LIST	006-D2767-02	3.00	Optional, Verify If Installed
GTX1	006-B0172-14	8.04	Optional, Verify If Installed (GTX 33D)
GTX1	006-B1607-BF	2.52	Optional, Verify If Installed (GTX 335)
GTX2	006-B1607-BF	2.52	Optional, Verify If Installed
GWX	006-B0266-12	2.20	
GWX FPGA	006-C0042-06	3.01	
MFD1	006-B1177-6H	20.90	
NAV1	006-B2253-00	2.00	
NAV2	006-B2253-00	2.00	

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.04 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version
PFD1	006-B1177-6H	20.90
PFD2	006-B1177-6H	20.90
PA-46-350P IMAGE	006-B2783-04	2783.04 (1-2)

(1) PIR-107997, Rev. B, factory installed in 4636769–4636775.
(2) See "Figure 52" on page 342502214.

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.06 (Sheet 1 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
COM1	006-B2371-02	2.02	
COM2	006-B2371-02	2.02	
GCU1	006-B0742-12	3.10	
GDC1	006-B1838-58	2.08	
GDC1 RGN LIST	006-D5080-50	2.00	
GDC2	006-B1838-58	2.08	
GDC2 RGN LIST	006-D5080-50	2.00	
GDL69	006-B1902-0B	5.51	Optional, Verify If Installed
GEA1	006-B2139-09	2.60	
GIA1	006-B2548-19	2.18	
GIA1 AUDIO	006-D0425-29	4.09	
GIA2	006-B2548-19	2.18	
GIA2 AUDIO	006-D0425-29	4.09	
GMA1	006-B2103-05	5.20C	
GMA1 AUDIO CONFIG	006-D3035-EP	5.21C	
GMA1 REGION LIST	006-D3035-D8	5.20C	
GMA1 AUX	006-B2104-05	5.20C	
GMA1 AUX ASR CONFIG 1	006-D3034-C4	5.50	
GMA1 AUX ASR CONFIG 2	006-D3034-08	2.00	
GMA1 AUX ASR CONFIG 3	006-D3034-09	2.00	
GMA1 AUX AUDIO DATABASE	006-D3034-36	4.20C	
GMA1 AUX REGION LIST	006-D3034-B7	5.20C	
GMC1	006-B0387-20	3.00	
GMU1	006-B0224-01	2.05	Optional, Verify If Installed
GMU1	006-B2679-00	2.00	Optional, Verify If Installed
GMU1 FPGA	006-C0048-00	2.00	
GMU2	006-B0224-01	2.05	Optional, Verify If Installed
GMU2	006-B2679-00	2.00	Optional, Verify If Installed
GMU2 FPGA	006-C0048-00	2.00	
GPS1	006-B1827-22	8.2	
GPS2	006-B1827-22	8.2	

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.06 (Sheet 2 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GRS1	006-B1838-09	2.09	
GRS1 REGION LIST	006-D5080-00	2.00	
GRS2	006-B1838-09	2.09	
GRS2 REGION LIST	006-D5080-00	2.00	
GSA PTCH CTL	006-B0398-46	3.56	
AFCS GAINS	006-D2018-03	2.03	
GSA PTCH MON	006-B0398-46	3.56	
AFCS GAINS	006-B2018-02	2.02	
GSA PTCH TRM C	006-B0398-46	3.56	
AFCS GAINS	006-D5863-02	2.02	
GSA PTCH TRM M	006-B0398-46	3.56	
AFCS GAINS	006-D5863-02	2.02	
GSA ROLL CTL	006-B0398-46	3.56	
AFCS GAINS	006-D2018-03	2.03	
GSA ROLL MON	006-B0398-46	3.56	
AFCS GAINS	006-D2018-03	2.03	
GSA YAW CTL	006-B0398-46	3.56	
AFCS GAINS	006-D2018-03	2.03	
GSA YAW MON	006-B0398-46	3.56	
AFCS GAINS	006-D2018-03	2.03	
GTS	006-B1244-61	4.01	Optional, Verify If Installed
GTS FPGA	006-C0081-21	2.10	Optional, Verify If Installed
GTS RGN LIST	006-D2767-02	3.00	Optional, Verify If Installed
GTX1	066-B0172-14	8.04	Optional, Verify If Installed (GTX 33D)
GTX1	006-B1607-0K	2.60	Optional, Verify If Installed (GTX 3X5)
GTX2	006-B1607-0K	2.660	Optional, Verify If Installed
GWX	006-B0266-12	2.20	Optional, Verify If Installed (GWX 68)
GWX	006-B2386-0A	2.55	Optional, Verify If Installed (GWX 75)

CHART 41
REQUIRED EQUIPMENT LISTS (REL)

G1000 NXi – VER. 2783.06 (Sheet 3 of 3)
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	
GWX FPGA	006-C0042-06	3.01	Optional, Verify If Installed (GWX 68)
MFD1	006-B1177-7Y	21.60	
NAV1	006-B2253-01	2.01	
NAV2	006-B2253-01	2.01	
PFD1	006-B1177-7Y	21.60	
PFD2	006-B1177-7Y	21.60	
PA-46-350P IMAGE	006-B2783-06	2783.06 (1-2)	

(1) PIR-107997, Rev. C, factory installed in 4636776 and up.
(2) See "Figure 52" on page 342502214.

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LANDING AND TAXI AIDS

Marker Beacon System

PIR-107825 NEW

A. Integrity Test

NOTE: Aeroflex IFR 4000 Nav/Com test set or equivalent is required for this test.

(1) Position the aircraft in an open area away from structures, personnel, and vehicles.

(2) Mount the test set antenna on a tripod and position with the following criteria:

- Aft of the aircraft's marker beacon antenna on aircraft centerline.
- Antenna tip is to be 36 inches from marker beacon antenna and 24 inches from the ground.
- Use a 15 foot coax cable so that test set is located in aircraft cabin during testing.

(3) Set test set parameters to the following:

- RF Frequency: 75 MHz
- Modulation: 95%
- Tone: 1300 Hz (Middle Marker)

When not using an Aeroflex IFR 4000 test set, follow the test equipment's instructions as provided by the manufacturer.

(4) Power up aircraft using battery power. Ensure that battery voltage is equal of greater than 22V.

(5) Starting at -50.0dBm, adjust tet set RF level until Middle Marker annunciation appears constant. Verify that marker audio signal is present over the pilot and co-pilot headphones and speaker. Verify that the Middle Marker appears at or below -17 dBm.

(6) If marker beacon annunciation is not present at or below -17 dBm, troubleshooting the marker beacon system is required.

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INDEPENDENT POSITION DETERMINING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Color Weather Radar (Optional)

A. General

Color Weather Radar is available as an option. When installed, the radar unit is mounted in a teardrop shaped pod hung from the right wing at the joint between the outboard and center wing sections (see "Figure 1"). A door in the right rear of the pod provides access to the void aft of the bulkhead, which can be used for miscellaneous lightweight storage, such as the ground protection equipment.

Maintenance instructions included in the manual are limited to description, removal, replacement, and self-test. Work exceeding this scope should be conducted by a qualified avionics repair facility in accordance with the component manufacturer's maintenance instructions.

(1) Allied Signal's Bendix/King RDR 2000 Vertical Profile Radar System

Available in S/N's 4636001 thru 4636459, 4636461 thru 4636462, and 4636481. See "Allied Signal's Bendix/King RDR 2000 Vertical Profile Radar System", below for more information.

(2) Garmin Airborne Weather Radar System (GWX 68 / 75)

GWX 68 is available in S/N's 4636460, 4636463–4636775; and, S/N's 4692134 and up, less 4692141, 4692149 and 4692153. GWX 75 is available in S/N's 4636776 and up.

See 34-25-01 or 34-25-02, as appropriate.

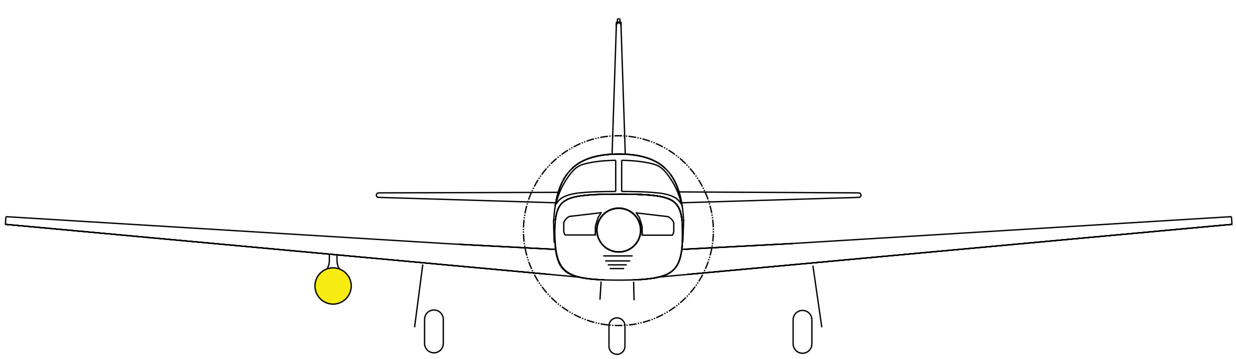
B. Allied Signal's Bendix/King RDR 2000 Vertical Profile Radar System

Available in S/N's 4636001 thru 4636459, 4636461 thru 4636462, and 4636481.

(1) Description

In S/N's 4636001 thru 4636374 only, it consists of the IN-182A radar display indicator in the center of the instrument panel and the ART 2000 antenna transceiver, AA201 OH antenna array, and CM 2000 configuration module mounted in the wing radar pod.

In S/N's 4636375 and up, the IN-182A display is replaced by the Avidyne Entegra MFD. See "Multifunction Display (MFD)" on page 34224, for setup and calibration.



Color Weather Radar Pod
Figure 1

The antenna transceiver and antenna array are assembled as a single unit and are mounted to the forward side of the bulkhead, inside the radar pod. The configuration module is separately mounted to the pod bulkhead.

The antenna transceiver and array assembly are covered by a removable radome secured to the circumference of the pod bulkhead with screws.

WARNING: AIRBORNE WEATHER RADAR SHOULD BE OPERATED ON THE GROUND ONLY BY QUALIFIED PERSONNEL.

WARNING: THIS INSTRUMENT GENERATES MICROWAVE RADIATION.

-- DO NOT ALLOW PERSONNEL TO STAND NEAR OR CROSS IN FRONT OF A TRANSMITTING RADAR ANTENNA.

-- DO NOT ALLOW PERSONNEL TO LOOK INTO A WAVEGUIDE, OR INTO THE OPEN END OF A COAXIAL CONNECTOR OR LINE CONNECTOR TO A RADAR TRANSMITTER OUTPUT.

WARNING: DO NOT OPERATE RADAR WHILE AIRCRAFT IS IN A HANGAR OR OTHER ENCLOSURE UNLESS THE RADAR TRANSMITTER IS NOT OPERATING OR THE ENERGY IS DIRECTED TOWARD AN ABSORPTION SHIELD WHICH DISSIPATES THE RADIO FREQUENCY (RF) ENERGY.

(2) Removal

(a) Remove screws (13) and remove the radome.

(b) Disconnect ground wires, antenna transceiver connector, and configuration module connector.

(c) Remove screws (4) and remove the antenna transceiver / array assembly.

(d) Remove screws (2) and remove the configuration module.

(3) Installation

(a) Position the configuration module and secure with screws (2).

(b) Position the antenna transceiver I array assembly arid secure with screws (4). Torque screws 60 to 70 inch pounds.

(c) Connect ground wires, antenna transceiver connector, and configuration module connector.

(d) Position the radome and secure with screws (13).

(4) Self-test (S/N's 4636001 thru 4636374 only.)

The following displays basic system fault information and tests antenna clearance. Use the GAIN, TILT, and function knobs and the RNG and mode buttons on the IN-182A indicator.

(a) Set the radar function knob to TST.

(b) Turn GAIN knob to minimum.

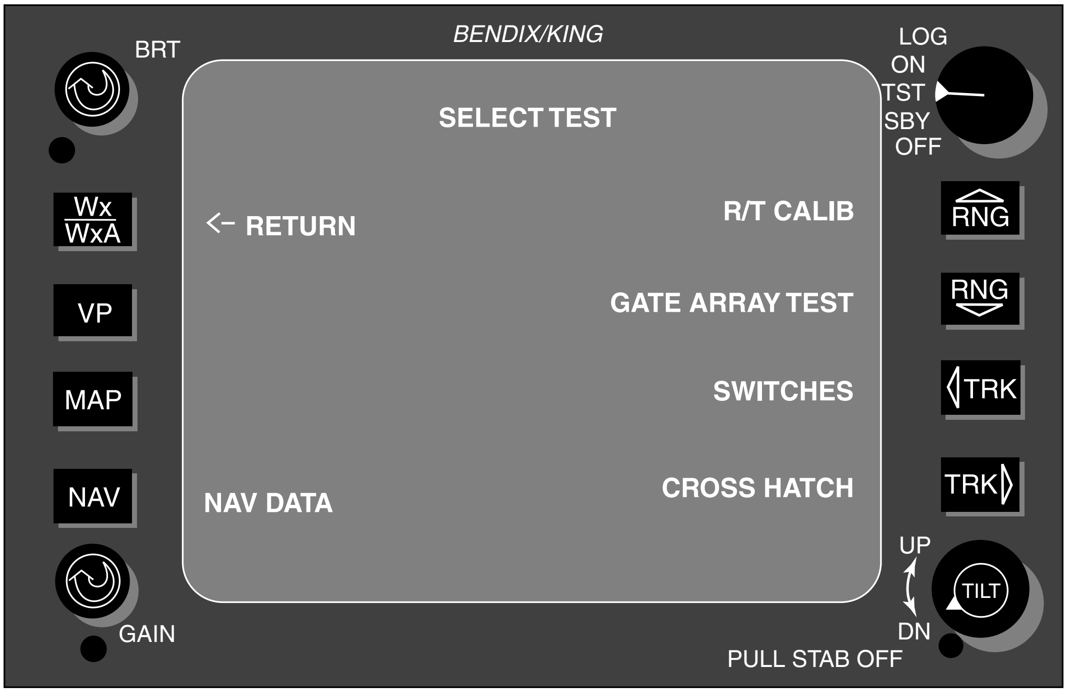
(c) Set antenna TILT to full up and select a range (RNG) of 240 NM.

(d) At the same time, depress the Wx/WxA and VP mode buttons to display the maintenance menu. (See Figure 2.)

(e) Press the range up (RNG ^) to display the RT CALIBRATION DATA page. (See Figure 3.)

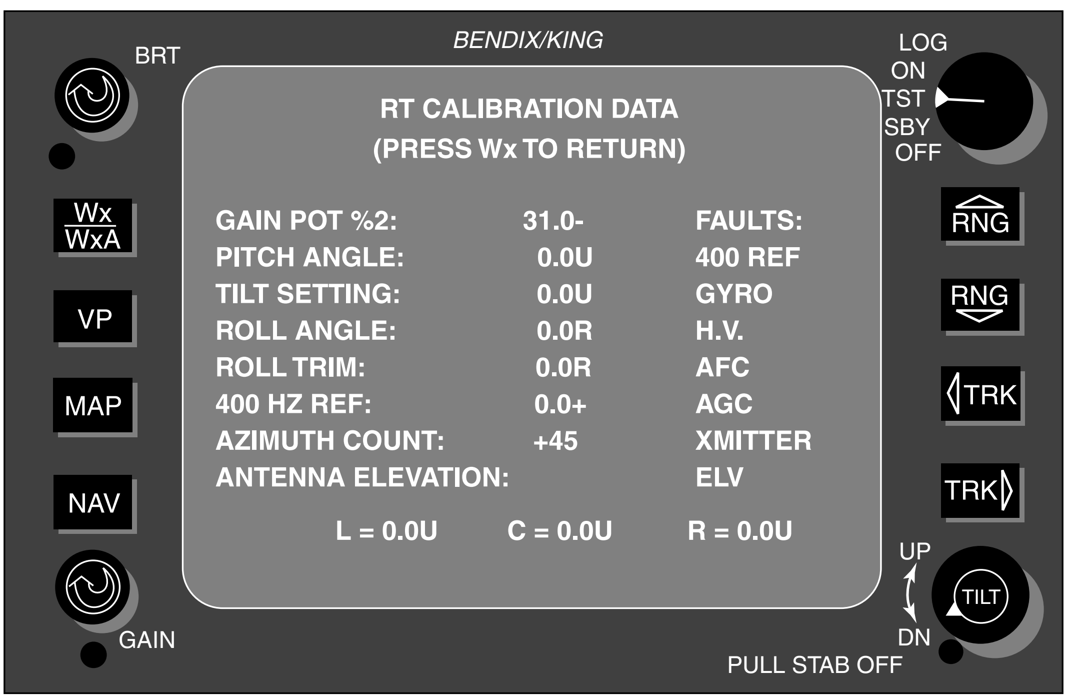
(f) As the RT CALIBRATION DATA page comes up, all fault fields will flash briefly. When the display stabilizes, the only fault field displayed should be XMITTER. If any other fault fields are displayed, further investigation is required.

(g) To initiate the antenna clearance test, rotate the GAIN knob to obtain a GAIN POT %2 value of -26.5 to -28. The antenna will then move to each of its extreme positions to verify that there are no physical interference and to ensure that all scan motors are working properly.



IN-182A Maintenance Menu
Figure 2

Effectivity
4636001 thru 4636374



IN-182A RT Calibration Data Display
Figure 3

Effectivity
4636001 thru 4636374

2. Traffic Advisory System - Avidyne TAS-610 (PA-46R-350T only.)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The Avidyne TAS-610 Traffic Advisory System is installed as optional equipment in S/N's 4692001 & up.

Information provided in this manual is intended solely to provide procedures to validate the set-up, operation and functionality of the TAS-610 Traffic Advisory System.

A. Removal and Installation

See 39-10-00.

B. Post Installation Setup Procedure

(PIR-PPS60241, Rev. A.)

(1) Aircraft Preparation and Configuration

Setup and preliminary checks may be made using the aircraft battery or external power (preferred) supplied by an external power cart.

(2) Computer Preparation and TAS Configuration

(a) Remove P6103 connector from TAS 610 processor.

(b) With power OFF on laptop computer and TAS 610, connect the laptop COM 1 serial port to the TAS 610 processor.

(c) Turn "ON" the laptop computer and start the HyperTerminal Application on the laptop.

(d) When the "new connection" dialog box appears asking for a connection name, type "TAS" and click "OK".

(e) When the "Connect to" dialog box appears, select the "Connect using" field to "Com 1" and click "OK".

(f) When the "COM 1 Properties" dialog box appears, set "Bits per second" to 9600. Verify that "Data bits" is set to 8, "parity" is set to "None"; click "OK".

(g) In the "TAS – HyperTerminal" window, select the File-> Properties menu item.

(h) When the "TAS Properties" dialog box appears, click the "Settings" tab at the top.

(i) Click "ON" the "ASCII Set-up.." at the bottom.

(j) On the "ASCII Set-up" dialog box, verify that "Echo typed characters locally" is checked and "character delay" is zero. Neither "send line ends with line feeds" nor "append line feeds to incoming line ends" should be checked. Ignore the other selections. Click "OK".

(k) Click "OK" on the "TAS properties" dialog box.

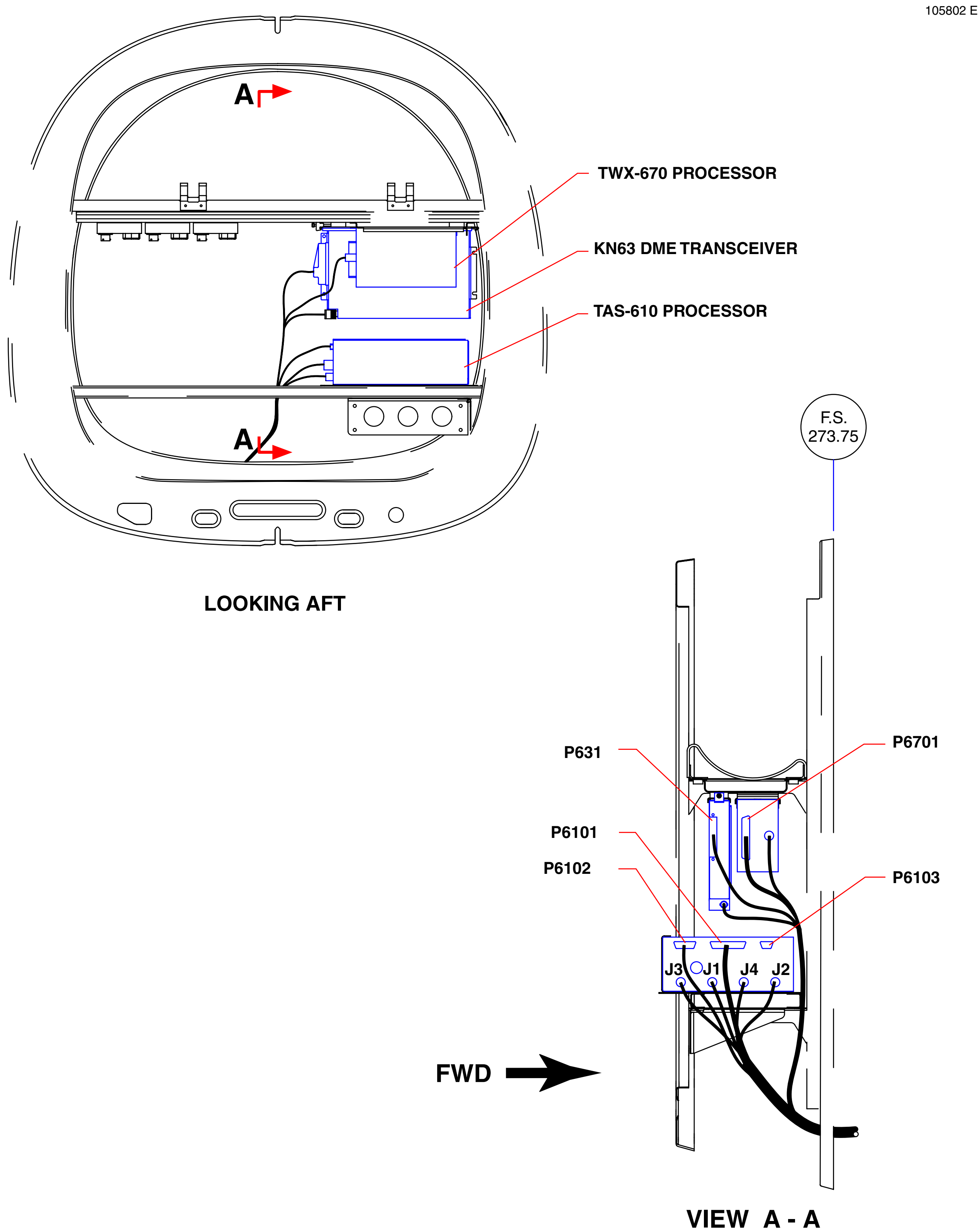
(l) Verify cable is still connected between laptop and TAS 610 processor. Turn on the Radio Master switch.

(m) In the "TAS-HyperTerminal" window, the message "Enter link message to connect:" should be displayed (this message may take up to 20 seconds to appear after starting the TAS 610 and HyperTerminal. If the message does not appear, check cable connections and Recycle power to the TAS 610 using the circuit breaker.

(n) When the "Enter link message to connect:" is displayed, type "CVI".

(o) The TAS 610 will administer a small math test. Add the first two numbers, and then add 3 to the numbers three times. For example, if the system says add 12 to 0, type into HyperTerminal dialog box: 12, 15, 18, 21, continue even if some of the numbers disappear and include the commas.

(p) A menu should appear. If the menu does not appear, the test was done incorrectly, press enter and start from step (m).



Aft Equipment Shelf
Figure 4

Effectivity
PA-46R-350T with Avidyne

(q) Press C to enter the Coupler Verification Test. Press E to enable test and verify test is PASSED, press X to return to the main menu.

(r) Press G for the Gear-Down verification test. Verify Gear-Down indication is displayed. Press X to return to the main menu.

(s) Press S for the Suppression Bus Verification test. Interrogate transponder using transponder test set. Press E to enable testing. Verify the "Toggle Random suppression" shows [MAN] mode. Press T to toggle Suppression on and off. Verify the percent replay of the host transponder reduces when Toggle Suppression shows [ON] and 100% when Toggle Suppression is [OFF]. Press X to return to the main menu.

(t) Press U for the audio setting screen. Press the + button to increase the volume; set the volume to 511. Press S and verify "testing" is audible through aircraft speaker. Press X to return to main menu.

(u) Press W for Weight on Wheels Verification test. Verify Weight on Wheels indication is "On Ground". Pull Traffic System circuit and disconnect the laptop cable from the TAS 610 processor. Install P6103.

3. Tactical Weather Detection System - Avidyne TWX-670 (PA-46R-350T only.)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The Avidyne TWX-670 Tactical Weather Detection System is installed as optional equipment in S/N's 4692001 & up.

Information provided in this manual is intended solely to provide procedures to validate the set-up, operation and functionality of the TWX-670 Tactical Weather Detection System.

A. Removal and Installation

See 39-10-00.

B. Post Installation Setup Procedure

(PIR-PPS60240, Rev. B.)

(1) Aircraft Preparation and Configuration

Setup and preliminary checks may be made using the aircraft battery or external power (preferred) supplied by an external power cart.

(2) Computer Preparation and TWX Configuration

(a) With power removed from the TWX 670 and laptop power "OFF", connect the laptop computer COM 1 serial port to the TWX 670 using a serial data loader cable.

(b) Turn "ON" the laptop computer and select the HyperTerminal application.

(c) When the "New Connection" dialog box is displayed, enter "TWX" (for connection name) and click "OK".

(d) When the "Connect To" dialog is displayed, select the "Connect Using" field to "COM 1" and click "OK".

(e) When the “COM 1 Properties” dialog box appears, set “Bits per Second” to 115,200. Verify “Data Bits” is set at “8”, “Parity” is set to “None”, “Stop Bits” is set to “1”, and “Flow Control” is set to “None”. Click “OK” to enter.

(f) Turn on Radio Master switch and verify that test data is scrolling down in the HyperTerminal window. Allow the boot sequence to complete, then press <ENTER>. Type “TWX” and press <ENTER>.

(g) Verify main menu is displayed, then press “C”. Verify the following settings are selected; use appropriate key to change setting if necessary.

Internal rate gyro installed:	=no
g - GPS configured:	=yes
h - Heading configured:	=yes
1 - Heading devise enabled	=yes
2 - GEO Stabilization	=yes
d - Audio enabled	=yes
t – Play Audio test sound	=100%

(h) Press ‘v’ to save configuration.

(i) Press ‘p’ to enter the port assignments menu and verify ports are assigned as shown:

RS-232 Ports:

Port 1:	DISPLAY	38,400 Baud
Port 2:	NO ASSIGNMENT	
Port 3	NO ASSIGNMENT	
Port 4	NO ASSIGNMENT	
Port 5	NO ASSIGNMENT	
Port 6	NO ASSIGNMENT	
Port 7	NO ASSIGNMENT	

ARINC 429 Ports	Equipment ID	Company ID	Speed
Port 1:	NONE	NONE	High
Port 2:	NO ASSIGNMENT		
Port 3:	Ø2	24	LOW
Port 4:	NO ASSIGNMENT		

(j) Press ‘x’ to return to the main menu. Remove power by pulling the WX DET SYSTEM circuit breaker. Disconnect the data loader cable and reset the WX DET SYSTEM circuit breaker.

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DEPENDENT POSITION DETERMINING

1. COM/NAV/GPS (GNS 430(W)/530)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Dual Garmin GNS-430 (GNS-430W in later airplanes) or Dual Garmin GNS-530 COM/NAV/GPS systems are installed as standard equipment in S/N's 4636248 & up. Each installation consists of the GNS-430(W) or GNS-530 transceiver/display units, associated wiring, and antennas.

A. Maintenance

Maintenance of the GNS units is “on condition” only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the appropriate GARMIN maintenance manual.

Information provided in this manual is intended solely to aid the removal and installation of the GNS transceiver/display units, their associated wiring and antennas.

B. Removal and Installation

See 39-10-00.

C. Post-Installation Lighting Set-up

See 33-10-00, Post-Installation Lighting Set-up.

D. Post Installation Setup Procedure

(PIR-PPS60199-2, Rev. K. / PPS60199-3, Rev. C. / PPS60199-7, Rev. New.)

(1) Aircraft Preparation and Configuration

Setup and preliminary checks may be made using the aircraft battery or external power supplied by an external power cart. Final checks (OBS/CDI or HSI accuracy) should be made with the engine running and the aircraft configured as in Chart 1.

(2) Configuration Setup Procedure

Access the Configuration Mode of the unit by depressing and holding the ENT key while applying power to the unit. Release the ENT key when the display activates. After the data base pages, press the ENT key twice (to step through the Data Base Information and Indicator Self Test Pages) to display the MAIN ARINC 429 CON FIG page. Pages may be selected by ensuring the cursor is off and rotating the small (inner) concentric knob on the right side of the unit.

To change data on the displayed Configuration Page, the cursor must be selected. Press the inner concentric knob on the right side of the unit to activate the cursor. Rotating the outer concentric knob on the right side of the unit changes the selected data field. Rotating the inner concentric knob changes the data within the selected field. To accept entry of the desired selection, press the ENT key.

CHART 1
FINAL CHECK CONFIGURATION

Engine running, alternators	On	Avionics	On
Instrument Panel Lights	Full Bright	Day/Night Switch	Day
Navigation Lights	On	Strobe Lights	On
Pitot Heat	Off	Air Conditioning	Off
Cabin Blower Fan	Low	Vent Blower	Off

(a) Main ARINC 429 Page

1) In PA-46-350P S/N's 4636248 and up, configure the MAIN ARINC 429 Page as follows:

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	HIGH	GARMIN GTX 330 or "AIR DATA" W/IHAS	HIGH	GARMIN GTX 330 or "AIR DATA" W/IHAS
IN 2	HIGH	OFF	HIGH	OFF
OUT	HIGH	GAMA 429 GRAPHICS W/INT	HIGH	OFF or GAMA 429 GRAPHICS W/INT with dual transponders
SDI	LNAV 1		LNAV 2	

2) In PA-46-350P S/N's 4636349 thru 4636374, with the optional Meggitt EFIS, configure the MAIN ARINC 429 Page as follows:

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	HIGH	GARMIN GTX 330 or "AIR DATA"W/IHAS	HIGH	GARMIN GTX 330 or "AIR DATA"W/IHAS
IN 2	HIGH	SANDEL EHSI	HIGH	SANDEL EHSI
OUT	HIGH	GAMA 429 GRAPHICS W/INT	HIGH	GAMA 429 GRAPHICS W/INT
SDI	LNAV 1		LNAV 2	

3) In PA-46-350P S/N's 4636375 and up, with Avidyne Entegra EFIS, configure the MAIN ARINC 429 Page as follows:

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	LOW	SANDEL EHSI	LOW	SANDEL EHSI
IN 2	HIGH	"EFIS/AIR DATA"	HIGH	"EFIS/AIR DATA"
OUT	LOW	GAMA 429 GRAPHICS	LOW	GAMA 429 GRAPHICS
SDI	LNAV 1		LNAV2	
VNAV*	ENABLE LABELS		ENABLE LABELS	

(* - only with GNS-430W)

4) In PA-46R-350Ts, configure the MAIN ARINC 429 Page as follows:

	GNS UNIT #1		GNS UNIT #2	
BUS	**SPEED**	**DATA**	**SPEED**	**DATA**
IN 1	LOW	SANDEL EHSI	LOW	SANDEL EHSI
IN 2	HIGH	"AIR DATA / AHRS"	HIGH	"AIR DATA / AHRS"
OUT	LOW	GAMA 429 GRAPHICS W/INT	LOW	GAMA 429 GRAPHICS W/INT
SDI	LNAV 1		LNAV2	
VNAV	ENABLE LABELS		ENABLE LABELS	

(b) Main RS-232 CONFIG Page

1) In PA-46-350P S/N's 4636248 and up; deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN RS232 CON FIG page. Configure as follows:

	GNS UNIT #1		**GNS UNIT #2**	
BUS	**INPUT**	**OUTPUT**	**INPUT**	**OUTPUT**
CHNL 1	OFF	"AVTN NO ALT"	OFF	OFF
CHNL 2	OFF	OFF or	OFF	OFF
	"HW EGPWS"			
	W/IHAS			
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	OFF	OFF	OFF	OFF
CHNL 5	OFF	OFF	OFF	OFF
CHNL 6	OFF	OFF	OFF	OFF

2) In PA-46-350P S/N's 4636349 thru 4636374, with the optional Meggitt EFIS; deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN RS232 CON FIG page. Configure as follows:

	GNS UNIT #1		**GNS UNIT #2**	
BUS	**INPUT**	**OUTPUT**	**INPUT**	**OUTPUT**
CHNL 1	OFF	"AVTN NO ALT"	OFF	OFF
CHNL 2	OFF	OFF or "HWEGPWS" W/IHAS	OFF	OFF
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	OFF	OFF	OFF	OFF
CHNL 5	OFF	OFF	OFF	OFF
CHNL 6	OFF	OFF	OFF	OFF

3) In PA-46-350P S/N's 4636375 and up, with Avidyne Entegra EFIS, deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN RS232 CONFIG page. Configure as follows:

	GNS UNIT#1		**GNS UNIT #2**	
BUS	**INPUT**	**OUTPUT**	**INPUT**	**OUTPUT**
CHNL 1	OFF	OFF	OFF	OFF
CHNL 2	OFF	OFF or "HW EGPWS" W/IHAS	OFF	OFF
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	OFF	OFF	OFF	OFF
CHNL 5*	OFF	OFF	OFF	OFF
CHNL 6*	OFF	OFF	OFF	OFF

(* - not with GNS-430W)

4) In PA-46R-350Ts, with Avidyne Entegra EFIS, deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN RS232 CONFIG page. Configure as follows:

	GNS UNIT #1		**GNS UNIT #2**	
BUS	**INPUT**	**OUTPUT**	**INPUT**	**OUTPUT**
CHNL 1	OFF	OFF	OFF	OFF
CHNL 2	OFF	OFF	OFF	OFF
CHNL 3	CROSSFILL	CROSSFILL	CROSSFILL	CROSSFILL
CHNL 4	OFF	OFF	OFF	OFF

(c) Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN System Config page. Configure as follows:

	GNS UNIT #1	GNS UNIT #2
CONFIGURE	FUEL	FUEL
FUEL TYPE	AVGAS	AVGAS
TERRAIN		
CONFIGURE	TERRAIN	TERRAIN
TERRAIN TYPE	TERRAIN or “TAWS” W/TAWS B	TERRAIN
TEST CARD?	PASS	PASS
HW CONFIGURE	TERRAIN or “TAWS”W/TAWS B	TERRAIN

(d) Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN LIGHTING page. Configure both the GNS #1 and #2 as follows:

	DISPLAY		KEY	
LIGHTING	NO SETTING		NO SETTING	
SOURCE	PHOTO		PHOTO	
RESP TIME/MIN	4	80	4	99
SLOPE/OFFSET	50	50	50	90
PHOTO TRANS % *	25			
PHOTO SLP/OFST *	35 **	35 **		

* Not with GNS-430W.

** Settings may vary unit-to-unit. Adjust ± 10 %, if needed, to obtain optimum lighting balance between panel units.

(e) Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the MAIN CDI/OBS CONFIG page. Setup is only required for selected course calibration. (GNS #1 only in GNS-430W installations.)

Adjust the OBS or HSI course needle to indicate a selected course of 150 degrees. The SELECTED COURSE field on the GNS unit should indicate very close to 150 degrees. Select the “Calibrate to 150?” field and press ENT. Verify the OBS (or HSI course) operation by checking that the course displayed on the GNS unit is withing 2° of the selected course. Verify the accuracy at 30° cardinal headings around the OBS card.

(f) Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the COM SETUP page. Change only the “SPACING” setting to display “Select 25.0 KHz.”

(g) Deselect the cursor and rotate the inner concentric knob on the right side of the unit to display the VOR/LOC/GS/CDI page. Change only the “DME CHNL MODE” setting to display “Parallel 2x5.”

(h) VOR/LOC/GS ARINC 429 CONFIG Page. Configure both the GNS #1 and #2 as follows:

1) without Avidyne Entegra:

GNS UNIT #1	**RX**	**TX**
SPEED	HIGH	HIGH
SDI	VOR/ILS 1	
DME MODE	DIRECTED FREQ 1	
GNS UNIT #2	**RX**	**TX**
SPEED	HIGH	HIGH
SDI	VOR/ILS 2	
DME MODE	DIRECTED FREQ 2	

2) with Avidyne Entegra:

GNS UNIT #1	**RX**	**TX**
SPEED	LOW	LOW
SDI	VOR/ILS 1	
DME MODE	DIRECTED FREQ 1	
GNS UNIT #2	**RX**	**TX**
SPEED	LOW	LOW
SDI	VOR/ILS 2	
DME MODE	DIRECTED FREQ 2	

(i) GPS antenna height is 6.5 feet.

E. VOR / LOC / GS Ramp Checks

Once the GNS units have been configured, proper navigation receiver and indicator operation (VOR / LOC / GS) should be verified using the IFR NAV-401L ramp test set or equivalent. This procedure assumes familiarity with proper operation of the test equipment and is not intended to be a tutorial on test set use. Refer to the appropriate test set operating instructions for proper operation of the ramp test equipment.

(1) OBS / CDI Checks:

(a) Check that the proper OBS indicator is connected to the appropriate VOR receiver; the #1 OBS is connected to GNS #1; and, the #2 OBS is connected to GNS #2.

(b) Check the lateral and vertical deviation needles and/or lateral and vertical flags for deflection while keying the communication transmitter on a representative sampling of frequencies (118.00 MHz to 136.975 MHz).

(c) Tune the navigation receiver and the adjust test set to 108.0 MHz. Using the ramp test set, check the accuracy of the OBS or Course Selector needle (HSI) at 30° increments (30°,60°, ... , 330°, 360°).

(2) Localizer Indicators Checks

(a) Tune the navigator receiver and adjust the test set to 108.1 MHz. Using the ramp test set, check that the course deviation needle (CDI) centers within 1/2 dot with 0 DDM applied at the test set.

(b) Check GNS #1 and #2 left and right deviation of the CDI as follows:

CDI (LOC)	DDM	DEFLECTION
Centered	0	0% F.S.
Standard Deviation	.093	60% F.S.
Full Scale Deflection	.155	100% F.S.
More than F.S.	.200	100% + F.S.
Full Tone (single mod)	.400	FLAG

(3) Glide Slope Indicator Checks

(a) Set navigation receiver and the test set to 108.1 MHz. Glide Slope generator defaults to proper G/S frequency (334.7 MHz.). Using the ramp test set, check that the needle (G/S) centers within 1/2 dot with 0 DDM applied at the test set.

(b) Check GNS #1 and #2 up and down deviation of the glide slope indicator as follows:

G / S	DDM	DEFLECTION
Centered	0	0% F.S.
Standard Deviation	.091	52% F.S.
Full Scale Deflection	.175	100% F.S.
More than F.S.	.400	100% + F.S.
Full Tone (single mod)	.800	FLAG

2. Transponder

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Information provided in this manual is intended solely to aid the removal and installation of the GTX transceiver/display units, its associated wiring and antennas.

A. S/N's 4636299, and 4636314 thru 4636348.

A Garmin GTX 327 Transponder is installed as standard equipment. Maintenance of the GTX 327 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the appropriate GARMIN GTX Maintenance Manual.

B. S/N's 4636349 thru 4636459.

A Garmin GTX 330 Transponder is installed as standard equipment. Maintenance of the GTX 330 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the appropriate GARMIN GTX Maintenance Manual.

C. S/N's 4636460, 4636463–4636715, 4636717–4636719; S/N's 4692134–4692140, 4692142–4692148, 4692150–4692152, 4692154–4692214.

A Garmin GTX 33 ES / GTX 33D ES Transponder is installed as standard equipment. A second unit may be installed as an option. Maintenance of the GTX 33 ES / GTX 33D ES is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the appropriate GARMIN GTX Maintenance Manual.

D. S/N's 4636720–4636775.

(1) A Garmin GTX 335R Transponder is installed as standard equipment.

(2) Optional configurations include:

(a) A single GTX 33D ES Transponder.

(b) Dual Garmin Transponders:

1) GTX 335R and GTX 345R.

2) GTX 33D ES and GTX 345R.

Maintenance of the these Garmin transponders is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the appropriate GARMIN GTX Maintenance Manual.

E. S/N's 4636776 and up.

(1) A Garmin GTX 345R Transponder is installed as standard equipment.

(2) Optional configurations include:

(a) A single GTX 345DR Transponder.

(b) Dual Garmin Transponders:

1) GTX 345R and GTX 335R.

2) GTX 345DR and GTX 335R.

Maintenance of the these Garmin transponders is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the appropriate GARMIN GTX Maintenance Manual.

F. Removal and Installation

See the appropriate transponder paragraphs under 'Components' in the appropriate section:

(1) S/N's 4636299, and 4636314 thru 4636348

See 39-10-00.

(2) S/N's 4636349 thru 4636459

See 39-10-00.

(3) S/N's 4636460, 4636463–4636715, 4636717–4636719;
S/N's 4692134–4692140, 4692142–4692148, 4692150–4692152, 4692154–4692214

See 34-20-00 and 39-10-00.

(4) S/N's 4636720–4636775

See 34-25-02 and 39-10-00.

(5) S/N's 4636776 and up

See 34-25-02 and 39-10-00.

G. Post Installation Setup Procedure

(1) S/N's 4636299, and 4636314 thru 4636348

See Chart 3, below.

(2) S/N's 4636349 thru 4636459

(PIR-PPS-60194; Rev. New / PPS-60206 Rev. O.)

Access the Configuration Mode of the unit by depressing and holding the FUNC key while applying power to the unit. Release the FUNC key when the display activates. The FUNC key sequences forward through the configuration pages. The START/STOP key reverses through the pages, stopping at the Menu page. The CRSR key highlights selectable fields on each page. When a field is highlighted, the 0-9 keys enter numeric data and the 8 or 9 keys move through list selections. Press the CRSR key to accept changes. When a field is highlighted, pressing the FUNC key moves to the next configuration page without saving the changes. To exit the configuration pages, turn the power off and then turn the unit on again without holding the FUNC key for normal operation.

NOTE: When the unit is turned on for the first time, or an invalid address is recognized, the unit will prompt the user to enter a valid aircraft address. See Mode S Address (ICAO Aircraft Address Code) and FLIGHT ID Entry Page paragraph, below.

(a) Configuration Menu

The JUMP TO menu page provides the capability to select a configuration mode starting page without having to step through all of the pages. Press the CRSR key and sequence through to the desired section with the 8 and 9 keys. Jump to the selection by pressing the CRSR key again with the desired selection highlighted. The FUNC key steps to the next configuration page, after which the START/STOP key reverses until stopping at the JUMP TO menu page. Following is a list of selections and their descriptions

SELECTION	DESCRIPTION
DIAGNOSTICS	Jumps to Gray Code Input Page
DISPLAY/AUDIO	Jumps to Audio Volume Page
I/O CONFIG	Jumps to ARINC INPUT #1 Page
ACFT CONFIG	Jumps to Operation Configuration #1 Page

(b) Refer to Charts 2 - 9 for required settings on the configuration pages.

(c) On the TEMPERATURE Page, select “NO” for SENSOR INSTALLED.

CHART 2
VOICE AND VOLUME
(TIS INSTALLATION ONLY - NO OTHER TRAFFIC SYSTEM INSTALLED.)

Function	Selection	Description
VOLUME	MAX	
VOICE	MALE	
MESSAGE (0-5)		Selected Audio Tones and Messages 0 = Toggle a continuous tone on and off 1 = Attention Tone, precedes voice messages to attract the pilot's attention. 2 = "Leaving Altitude", when altitude monitor is active and the altitude deviation is exceeded. 3 = "Traffic",when a TIS traffic alert is received. 4 = "Time Expired", when the countdown expires. 5 = 'Traffic Not Available", when TIS service is not available or out or range of an operating TIS MODE S site. 6-9 are not used.
ALTITUDE MONITOR	OFF TONE	When Altitude Pre-Select is installed. When Altitude Pre-Select is not installed.
PAGE CHANGE	ENABLE	
COUNTDOWN TIMER	TONE	

CHART 3
GTX-327/-330 DISPLAY MODE AND KEY LIGHTING

Data Element	Selection
DISPLAY MODE PAGE	
DISPLAY MODE	AUTO
LEVEL	75
DISPLAY BACKLIGHT PAGE	
BKLT	AUTO
LVL	Not Selectable
RSPTIME	4
MIN	8
BKLT SRCE	PHOTO
SLOPE	50
OFFSET	50
KEY LIGHTING PAGE	
KEY	AUTO
LVL	Not Selectable
RSPTIME	4
MIN	5
KEY SRCE	28V
SLOPE	20
OFFSET	30
CONTRAST MODE	AUTO
VFR KEY	ENABLE

CHART 4
ARINC 429 CONFIGURATION PAGE - STANDARD INSTALLATION
(DUAL TRANSPONDERS) (S/N'S 4636001 THRU 4636374)

	#1 GTX 330		#2 GTX 330	
429 INPUT	SPEED	DATA	SPEED	DATA
CHANNEL 1	HIGH	GPS/FMS	HIGH	GPS/FMS
CHANNEL 2	LOW	OFF	LOW	OFF
CHANNEL 3	LOW	OFF	LOW	OFF
CHANNEL 4		OFF		OFF
429 OUTPUT		DATA		DATA
CHANNEL 1		OFF		OFF
CHANNEL 2		GARMIN W/TIS GARMIN W/IHAS		GARMIN W/TIS GARMIN W/IHAS

CHART 5
ARINC 429 CONFIGURATION PAGE - MEGGITT EFIS OPTION
(DUAL TRANSPONDERS) (S/N'S 4636349 THRU 4636374)

	#1 GTX 330		**#2 GTX 330**	
429 INPUT	SPEED	DATA	SPEED	DATA
CHANNEL 1	HIGH	GPS/FMS	HIGH	GPS/FMS
CHANNEL 2	HIGH	ADC W/ALT	HIGH	ADC W/ALT
CHANNEL 3	LOW	OFF	LOW	OFF
CHANNEL 4		OFF		OFF
429 OUTPUT		DATA		DATA
CHANNEL 1		OFF		OFF
CHANNEL 2		GARMIN W/TIS GARMIN W/IHAS		GARMIN W/TIS GARMIN W/IHAS

CHART 6
ARINC 429 CONFIGURATION PAGE - AVIDYNE ENTEGRA
(DUAL TRANSPONDERS) (S/N'S 4636375 AND UP)

	#1 GTX 330		**#2 GTX 330**	
429 INPUT	SPEED	DATA	SPEED	DATA
CHANNEL 1	LOW	GPS/FMS	LOW	GPS/FMS
CHANNEL 2	HIGH	ADC W/ALT	HIGH	ADC W/ALT
CHANNEL 3	LOW	OFF	LOW	OFF
CHANNEL 4		OFF		OFF
429 OUTPUT		DATA		DATA
CHANNEL 1		OFF		OFF
CHANNEL 2		“OFF” W/IHAS “GARMIN W/TIS” W/O IHAS		“OFF” W/IHAS “GARMIN W/TIS” W/O IHAS

CHART 7
ARINC 429 CONFIGURATION PAGE - AVIDYNE ENTEGRA
(SINGLE TRANSPONDER) (S/N'S 4692001 AND UP)

	GTX 330	
429 INPUT	SPEED	DATA
CHANNEL 1	LOW	GPS/FMS
CHANNEL 2	HIGH	ADC W/ALT
CHANNEL 3	LOW	OFF
CHANNEL 4		OFF
429 OUTPUT		DATA
CHANNEL 1		OFF
CHANNEL 2		"GARMIN W/TIS" "GARMIN W/TAS 610"

CHART 8
RS-232 INPUT AND OUTPUT

RS-232 CONFIG	**GTX**	
232 INPUT	INPUT	OUTPUT
CHANNEL 1	OFF	OFF
CHANNEL 2	OFF	OFF

CHART 9
OPERATION CONFIGURATION

VS RATE	500
FORMAT	FEET
VFR ID	1200
ALTITUDE ALERT DEVIATION	200
SQUAT SWITCH	YES
SENSE	LOW
DELAY TIME	24
AUTO FLIGHT TIMER	MANUAL

(d) MODE S Address (ICAO Aircraft Address Code) and FLIGHT ID Entry Page

When the unit is turned on for the first time, or an invalid address is recognized, the unit will prompt the user to enter a valid aircraft address.

NOTE: U.S. registered aircraft ICAO Aircraft Address Code (ADDRESS HEX) is the "N number" and is displayed as such on the ATC-601. On Non-U.S. registered aircraft, verify only that the hexadecimal code displayed on ATC-601 matches the "ADDRESS HEX" code programmed on the GTX-330.

1) U.S. Registered Aircraft
   a) To highlight the "U.S. Tail #" address field, press the CRSR key one time.
   b) Enter the registration number using the number keys. Press a key repeatedly to scroll through the digit/alpha characters for that entry field.
   c) Press the CRSR key to select the numeric entry field. Enter the next character as stated in step b , then move onto the next one, repeating the process until the complete number is entered.
   d) When finished, press the CRSR key to accept the number entry.
   e) For entering the Flight ID number, press the CRSR key one time.
   f) Repeat steps b and c.
   g) When finished, press the CRSR key to accept the number entry.
2) Non-US Registered Aircraft
   a) For entering Non-U.S. Aircraft Registration, press the CRSR key one time, then 8 or 9 to select ADDRESS HEX.
   b) Non-U.S. Aircraft Registry normally supplies a 24 bit binary code, known as the ICAO Aircraft Code for the Mode S address in lieu of directly entering the N# for U.S. registered aircraft.
   c) The GTX 330 only accepts a hexadecimal format for the Non-U.S. Registered Aircraft, so the 24 bit binary code must be converted to a hexadecimal format.
   d) Convert the 24 bit binary code to hexadecimal as follows:
      1] 24 bit binary code: 111001000111111101101101
      2] Separate for hex conversion: 1110 / 0100 / 0111 / 1110 / 1110 / 1101
      3] Apply the hexadecimal values found in Chart 10.

E	4	7	E	E	D
1110	0100	0111	1110	1110	1101

      4] "ADDRESS HEX" = E47EED
   e) Enter the hexadecimal address using the number keys. Press a key repeatedly to scroll through the digit/alpha characters for that entry field.

CHART 10
HEXADECIMAL CONVERSION

Binary	Hexadecimal	Decimal	Binary	Hexadecimal	Decimal
/0000	0	0	/1000	8	8
/0001	1	1	/1001	9	9
/0010	2	2	/1010	A	10
/0011	3	3	/1011	B	11
/0100	4	4	/1100	C	12
/0101	5	5	/1101	D	13
/0110	6	6	/1110	E	14
/0111	7	7	/1111	F	15

f) Press the CRSR key to select the numeric entry field. Enter the next character as stated in step e , then move onto the next one, repeating the process until the complete number is entered.

g) For FLIGHT ID select “CON FIG ENTRY” and enter the Hex code or select “POWER UP ENTRY” .

h) When finished, press the CRSR key to accept the number entry.

(e) On the “MODE S Aircraft Type” Page, select “<15.5k Lb” for AC TYPE, “<= 300 kt” for MAX AIRSPEED, and “1E-5” for GPS INTEGRITY. Continue on and select “<15 MT” for AC LENGTH TYPE LENGTH, <=11.5 MT. FOR AC WIDTH TYPE WIDTH, AND “ENABLE” for EHS (ENHANCED SURVEILLANCE).

NOTE: Advance through remaining display screens that cannot be changed.

(f) When entries are complete, cycle unit power and check for entries.

(3) S/N’s 4636460, 4636463–4636715, 4636717–4636719;
S/N’s 4692134–4692140, 4692142–4692148, 4692150–4692152, 4692154–4692214
(Garmin G1000)

See 33-10-00, Post-Installation Lighting Set-up.

(4) S/N's 4636720–4636775 (Garmin G1000 NXi)

See "GTX 3X5R Mode S/ES Transponder" on page 342502101 for Removal and Installation.

(5) S/N’s 4636776 and up (Garmin G1000 NXi)

See "GTX 3X5R Mode S/ES Transponder" on page 342502101 for Removal and Installation.

H. Mode S Transponder Ground Test

(PIR-PPS60207, Rev. F.)

NOTE: Provides compliance with FAR Part 43 Appendix F and Part 91.413, and also tests for accuracy, correlation requirements for altitude encoders to provide compliance with FAR Part 43 Appendix E.

These tests enable compliance with FAR 91.413 and FAR Part 43 Appendixes E and F. Perform these tests in conjunction with Post Installation Set-up, above.

(1) Requirements

(a) Tests to be accomplished by using a Transponder Ramp Tester capable of testing to requirements stated in FAR Part 43 Appendix F.

(b) Transponder must be properly configured per aircraft setup specification prior to starting this process.

(c) Controllable vacuum source.

(d) During all tests specified herein, the aircraft altimeter shall be set a 29.92 inches of mercury (1013.2 Millibars).

(e) Altitude encoders shall be tested up to and including the maximum operating altitude for that aircraft.

(2) Auto Test Requirements

(a) Perform the AUTO TEST function and verify that MODES A, C, and S PASS.

(b) Record the following results:

1) FREQ: 1090 +/-1 MHz (1089.0 MHz to 1091.0 MHz)

2) ERP: 125 Watts minimum to 500 Watts maximum

3) MTL: -73 dBm +/- 4 (-69 dBm to -77 dBm)

4) DIVERSITY ISOLATION: Greater than 20db (Only on aircraft equipped with diversity transponder.)

(3) Individual Test Requirements

U.S. Registered Aircraft ICAO aircraft address code (address hex) is actually the "N number". On foreign registered aircraft verify only that the hexadecimal code displayed matches the "address hex" code programmed on the transponder.

(a) SLS Level

NOTE: This can be verified via ATC 601 TEST 4 or IFR 6000 A/C DECDR/SLS.

(b) Test Set interrogates UUT with ATCRBS Modes A and C. It verifies a reply is received when P2 pulse is 9dB less than P1 pulse and no reply is received when P2 pulse amplitude is equal to P1 pulse. Transponder response to Mode 3/A interrogations not to exceed 1% when amplitude of the P2 pulse = P1 pulse and the transponder is interrogated at a pulse repetition rate of 235 per second.

(c) ATCRBS Only All-Call

NOTE: This can be verified via ATC 601 TEST 5 or IFR 6000 A/C F1/F2 SPACE/WIDTH.

Test Set interrogates UUT with ATCRBS Only ALL-CALL (0.8 microsecond P4 pulse) and verifies that Mode S transponders do not reply.

(d) MODE S All-Call Interrogations

NOTE: This can be verified via ATC 601 TEST 6 or IFR 6000 S. ALL-CALL.

Test Set interrogates UUT with Mode A/S All-Call (1.6 microsecond P4 pulse). Address received from Mode S reply is sent in UF4 interrogation and verified with DF4 address.

(e) Invalid Address: Mode S Address

NOTE: This can be verified via ATC 601 TEST 7 or IFR 6000 S REPLY.

Test Set interrogates UUT with MODE S interrogations using correct address and two incorrect addresses (Nominal rate of 50 interrogations per second). These addresses are different from the address determined from Mode S All-Call. The Test Set verifies that no reply is received.

(f) Mode S UF0:

NOTE: This can be verified via ATC 601 TEST 9 or IFR 6000 UF0.

Test Set interrogates UUT with Mode S UF0 and verifies that reply has correct altitude (Compared to ATCRBS Mode C), correct address (Compared to DF11) and correct format. UFO AQ bit is "1" for requesting airspeed information in DFO RI field.

(g) Mode S UF4:

NOTE: This can be verified via ATC 601 TEST 10 or IFR 6000 UF4.

Test Set interrogates UUT with Mode S UF4 and verifies reply has correct altitude (Compared to ATCRBS Mode C), correct address (Compared to DF11) and correct format.

(h) Mode S UF5:

VVNOTE: This can be verified via ATC 601 TEST 11 or IFR 6000 UF5.

Test Set interrogates UUT with Mode S UF5 and verifies that reply has correct ID Code (Compared to ATCRBS mode A), correct address (Compared to DF11) and correct format.

(i) Mode S UF11:

NOTE: This can be verified via ATC 601 TEST 12 or IFR 6000 UF11.

Test Set interrogates UUT with Mode S UF16 and verifies that reply has correct address and format.

(j) Mode S UF16:

NOTE: This can be verified via ATC 601 TEST 13 or IFR 6000 UF16.

Test set interrogates UUT with UF16 and verifies that reply has correct altitude (Compared to ATCRBS Mode C), correct address (Compared to DF11) and correct format. UF16 AQ bit is "0" for requesting TCAS capability information in DF16 RI field.

(k) Mode S UF20:

NOTE: This can be verified via ATC 601 TEST 14 or IFR 6000 UF20.

Test Set interrogates UUT with Mode S UF20 and verifies that reply (if received) has correct ID code (Compared to ATCRBS Mode A), correct address (Compared to DF11) and correct format.

(l) Mode S UF21:

NOTE: This can be verified via ATC 601 TEST 15 or IFR 6000 UF21.

Test Set interrogates UUT with Mode S UF21 and verifies that reply (if received) has correct ID code (Compared to ATCRBS Mode A), correct address (Compared to DF11) and correct format.

(m) Squitter:

NOTE: This can be verified via ATC 601 TEST 15 or IFR 6000 S REPLY.

Test Set verifies that squitters are received from UUT every 0.8 to 2.4 seconds.

(n) Radio Reply Frequency

NOTE: This can be verified via ATC 601 TEST 17 or IFR 6000 POWER/FREQ.

Test Set verifies that UUT transmit frequency is 1090 MHz (+/- 1 MHz).

(o) Mode S Diversity Transmission Channel Isolation:

NOTE: This can be verified via ATC 601 TEST 18 or IFR 6000 S REPLY. (Test only applicable to aircraft equipped with Diversity Transponder)

Test set verifies that power level difference is 20dB between "on " antenna squitters and "off" antenna squitters.

NOTE: To insure $\geq$ 20dB dynamic range, Diversity Test must be run within 50 ft of UUT Antenna being tested.

(p) MTL Difference (Receiver Sensitivity)

NOTE: This can be verified via ATC 601 TEST 19 or IFR 6000 POWER/FREQ.

Test Set verifies that receiver sensitivity difference between Mode A and Mode C is $\leq$ 1.0 dB.

(q) Power and MTL:

NOTE: This can be verified by ATC 601 Power Test or IFR 6000 POWER/FREQ.

(r) Test Set verifies that UUT power output is between the following parameters:

- 48.5 dBm < UUT ERP < 57.0 dBm
- UUT MTL = -73.0 dBm (+/- 4.0 dB)

NOTE: Power Test must be run at a distance established in Setup procedure. The minimum RF peak output power (ERP on Test Set) is at least 125 watts and maximum RF peak output power does not exceed 500 watts.

(4) Accuracy Test

(a) Set-up

1) Apply power to the Altitude Reporter and the Transponder Test Set.

2) Set primary flight altimeter to 29.92 in hg.

3) Allow a minimum of 15 minutes warm up time.

4) The Transponder Test Set reading should match with flight altimeter reading, not with the static pressure system tester.

(b) Testing

Select ten or more evenly spaced altitude test points between sea level and the altitude designated in 3.6. Increase the vacuum, causing the aircraft altimeter indicated altitude to increase. Approach each test point slowly. Tap the altimeter lightly before each reading to minimize friction effects. If a check point is missed, do not go back, go on to the next test point. For each test point check the aircraft altimeter dial reading at the encoding altimeter/ digitizer transition point. (The transition point is the point at which the altitude encoder readout displayed on the ATC-600A or equivalent test set, changes to the proper code for the test point).

(c) Accuracy Requirements

Accuracy requirements shall be as follows: The altitude encoder shall transition to the selected test point within ±125 ft. at each test point, and within ±87 ft. at no less than 70 percent of the test points.

EXAMPLE: Approaching the 6,000 foot transition point, the aircraft altimeter dial shall read 6,000 ±125 ft. (5,875 to 6,125 ft.) when the transition to the ±87 ft. (5,913 to 6,087 ft.) if the 6,000 foot transition point is one of 70% noted above).

(d) Out of Limits

If either the aircraft altimeter of the aircraft encoder fails the accuracy tests, replace the out-of-limits device and repeat the test.

(e) Post Test Cleanup

After testing is completed, the aircraft transponder switch and battery switch shall be placed in the off position, the test set shall be disconnected. The static system shall be reconnected, and the static system then tested for leaks.

(f) ADS-B Out Functionality Tests

Perform Advisory Circular 20-165 test and verify the following:

1) NACp ≥ 8

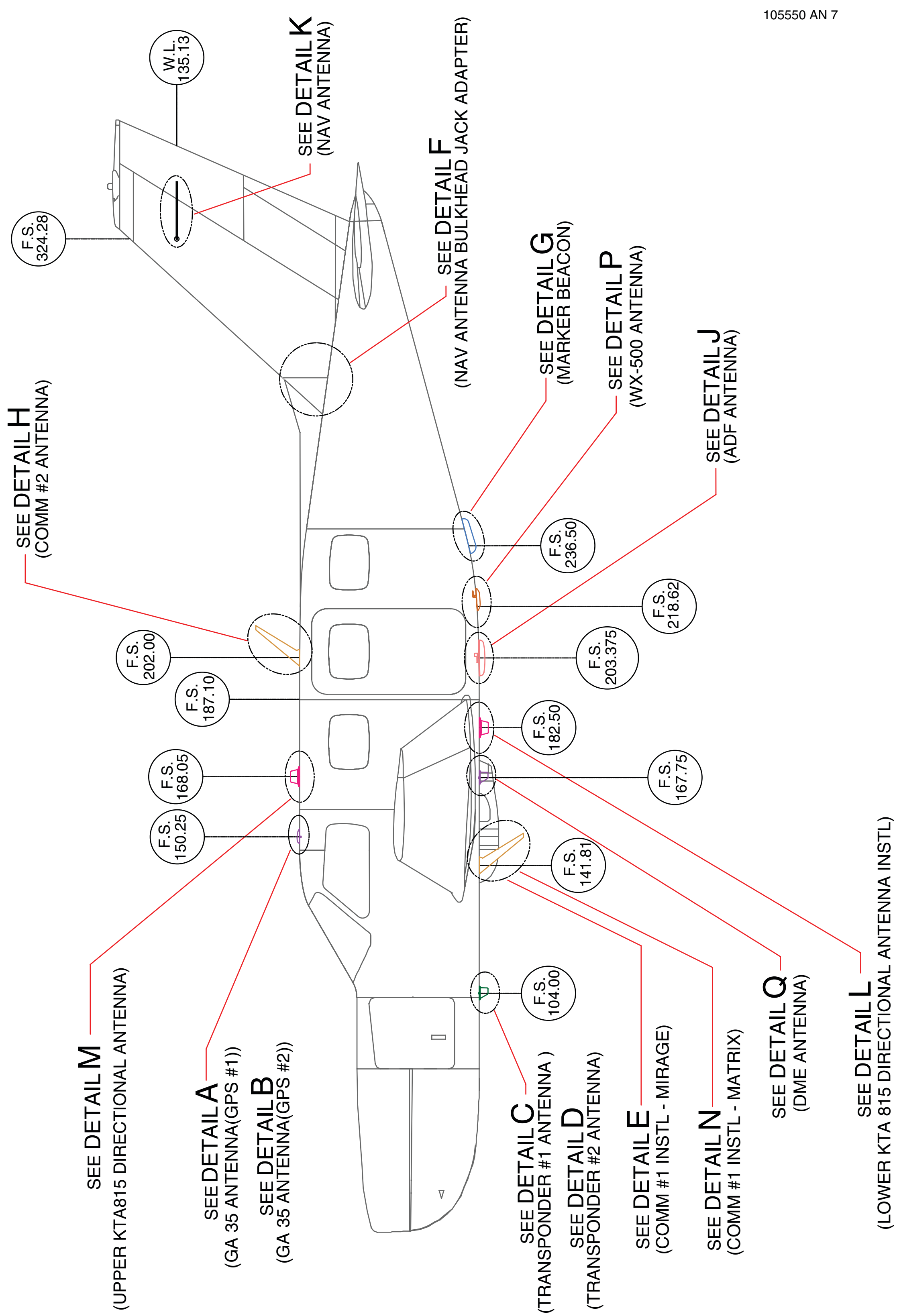
2) NIC ≥ 7

3) NACv ≥ 1

4) Verify ADS-B barometer altitude matches transponder barometer altitude (Alt. Air= 0).

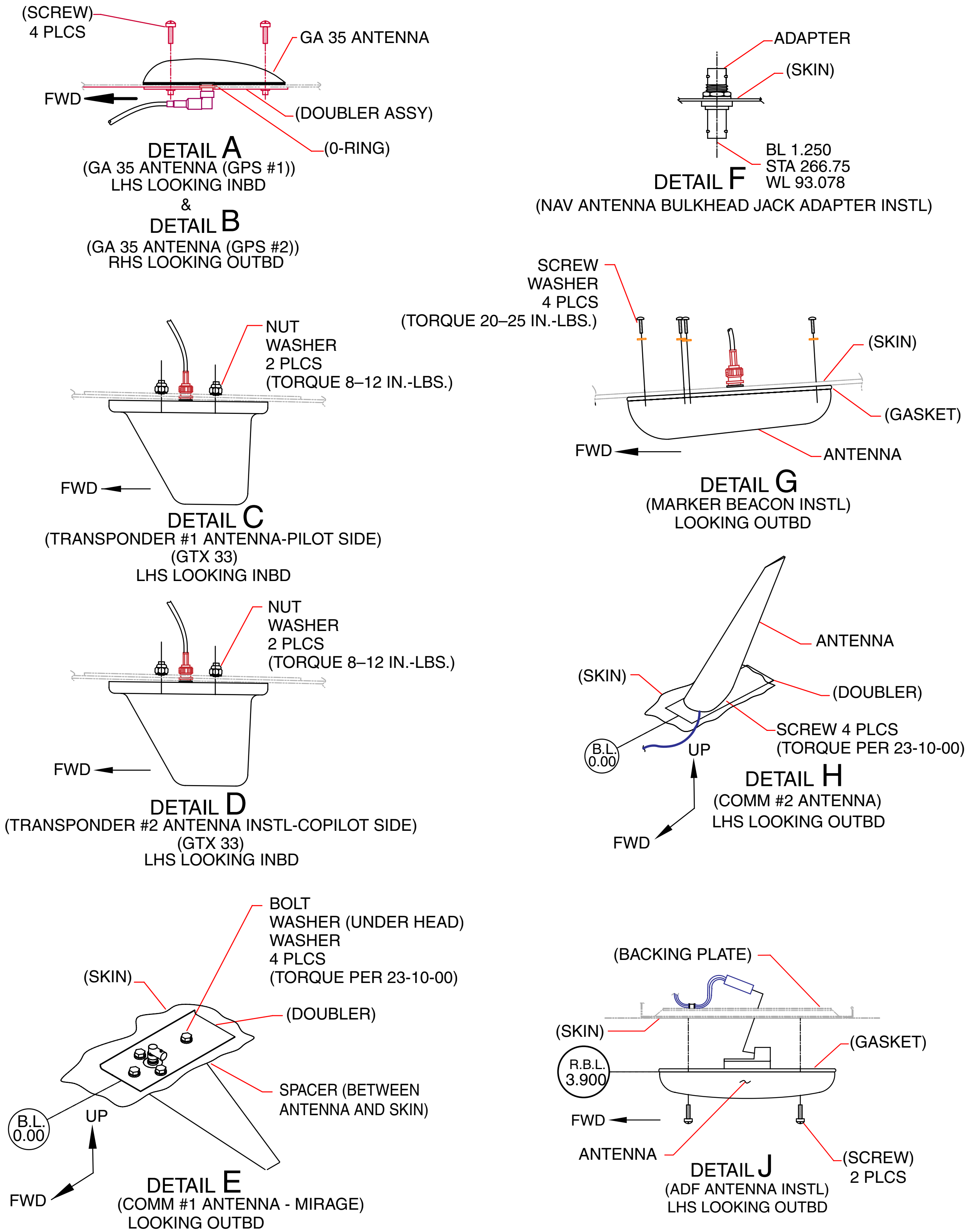
(5) Test Complete.

Restore airplane to operational configuration and secure.



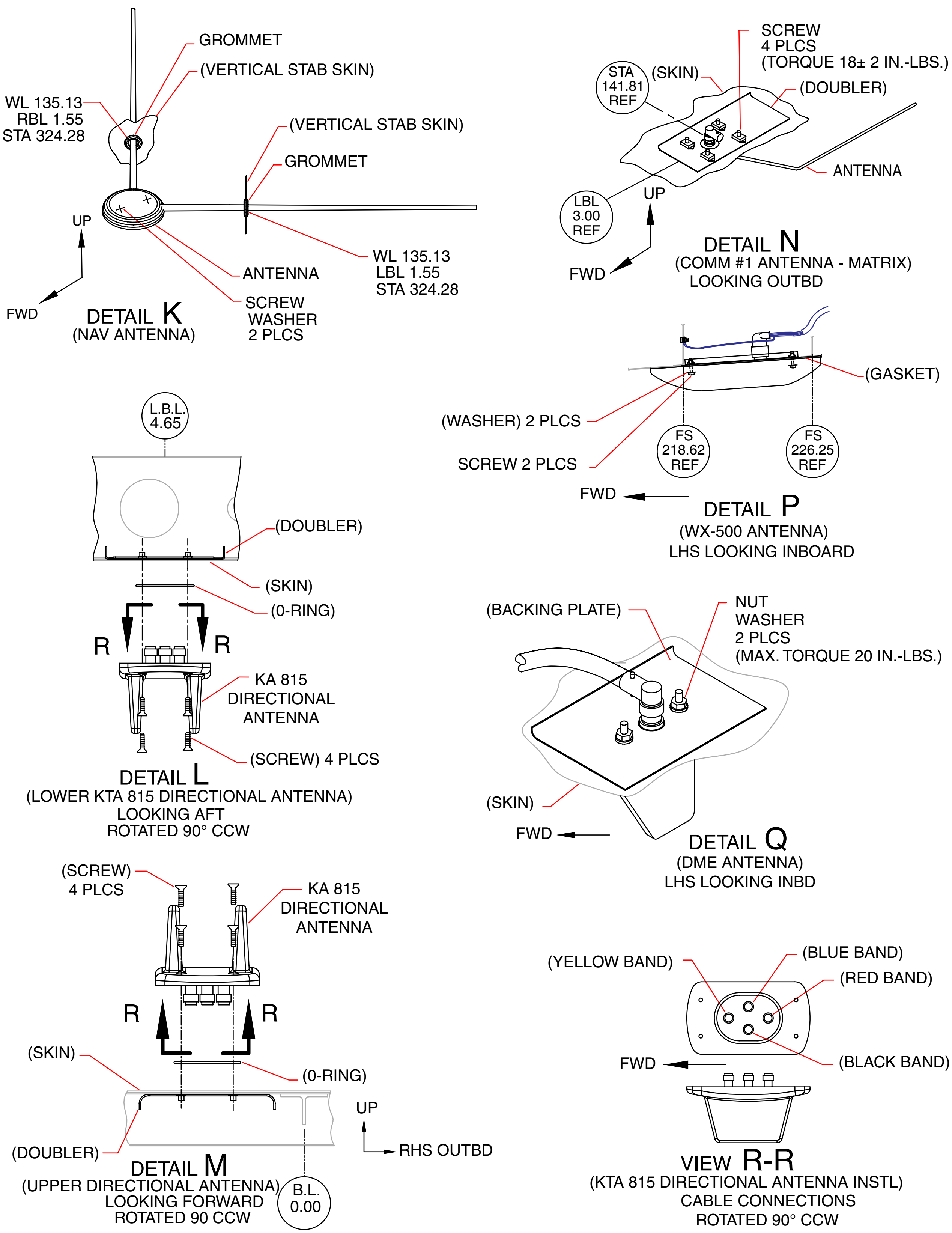
Effectivity
with Garmin G1000

Antenna Installation
Figure 1 (Sheet 1 of 3)



Antenna Installation
Figure 1 (Sheet 2 of 3)

Effectivity
with Garmin G1000



Effectivity
with Garmin G1000

Antenna Installation
Figure 1 (Sheet 3 of 3)

3. RA-3502 ADF

See "Figure 2".

Maintenance instructions included in the manual are limited to description, removal, replacement, and self-test. Work exceeding this scope should be conducted by a qualified avionics repair facility in accordance with the component manufacturer's maintenance instructions.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. Description

The ADF Receiver AD 3502 is designed as a single-block unit. Its dimensions correspond to the ARINC standard for control equipment. All controls, indicators and displays are located on the front plate. The back side of the unit contains: the unit connector plug and the antenna jack.

RA-3502 - ADF RECEIVER-REMOTE

AN-3500 ANTENNA

RA-3502 ADF
Figure 2

B. Troubleshooting

Refer to RA-3502 ADF Maintenance Manual.

C. Removal

See "Figure 3" on page 345023.

(1) RA-3502 ADF
   (a) Master power off.
   (b) Remove the aft cabin hatshelf assembly.
   (c) Remove the cabin rear closeout panel to gain access to the aft avionics shelf.
   (d) Loosen quick release bolts and slide ADF receiver out.
   (e) Disconnect harness assembly and antenna cable from ADF receiver.
   (f) Remove ADF receiver.

(2) AN3500 Antenna
   (a) Master power off.
   (b) Remove screws securing antenna to fuselage.
   (c) Disconnect cable from antenna.
   (d) Remove antenna.

D. Installation

See "Figure 3" on page 345023.

(1) AN3500 Antenna
   (a) Position antenna fuselage.
   (b) Attach cable to antenna.
   (c) Secure with screws.

(2) RA-3502 ADF
   (a) Position ADF receiver on the aft avionics shelf.
   (b) Attach harness assembly and antenna cable to ADF receiver and slide into mount.
   (c) Tighten quick release bolts.
   (d) Replace the cabin rear closeout panel.
   (e) Replace the aft cabin hatshelf assembly.

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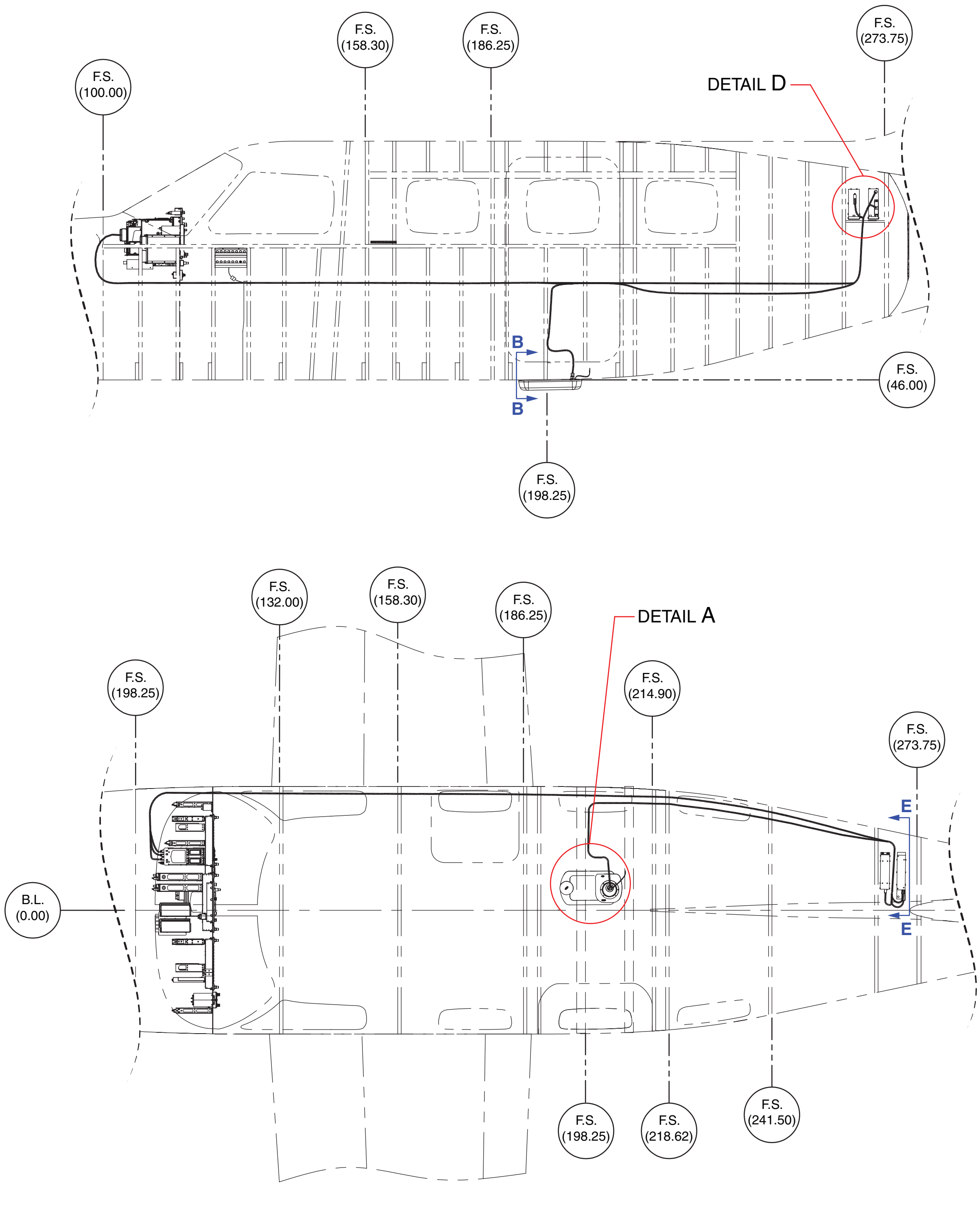
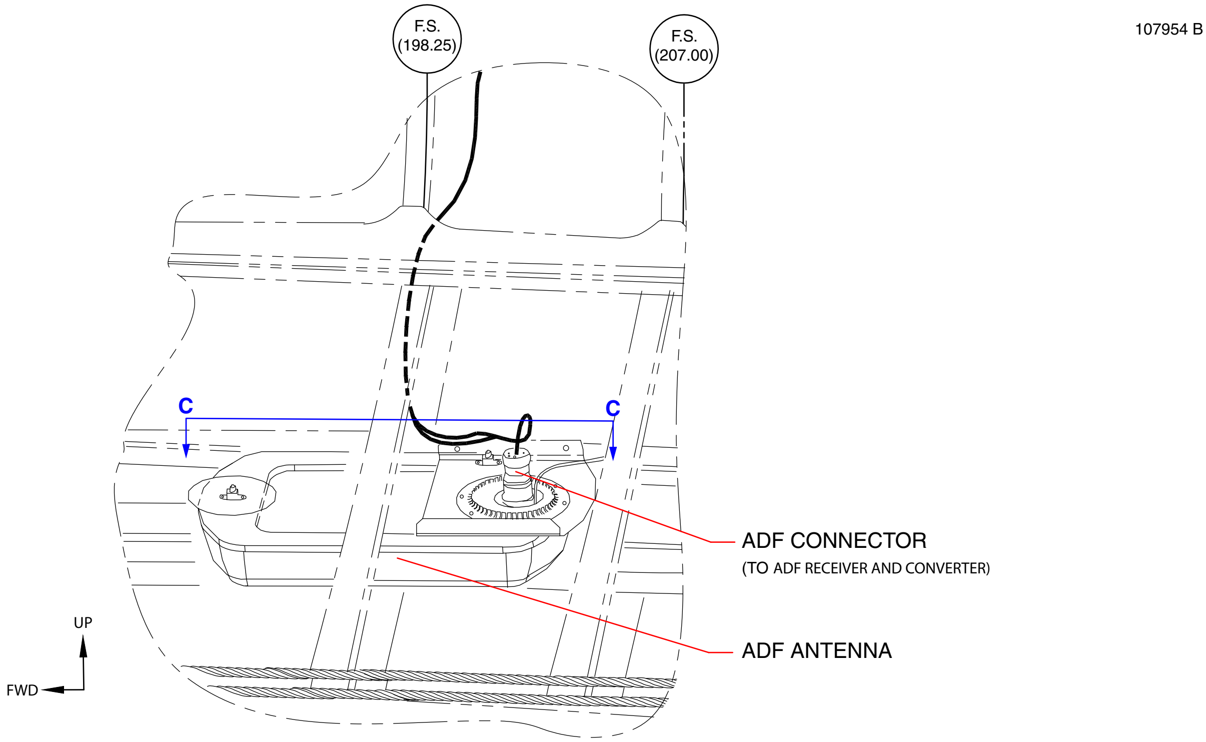
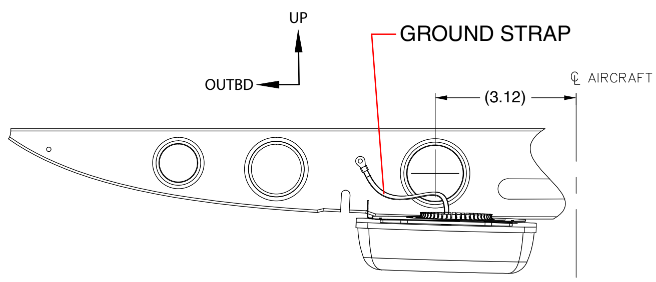


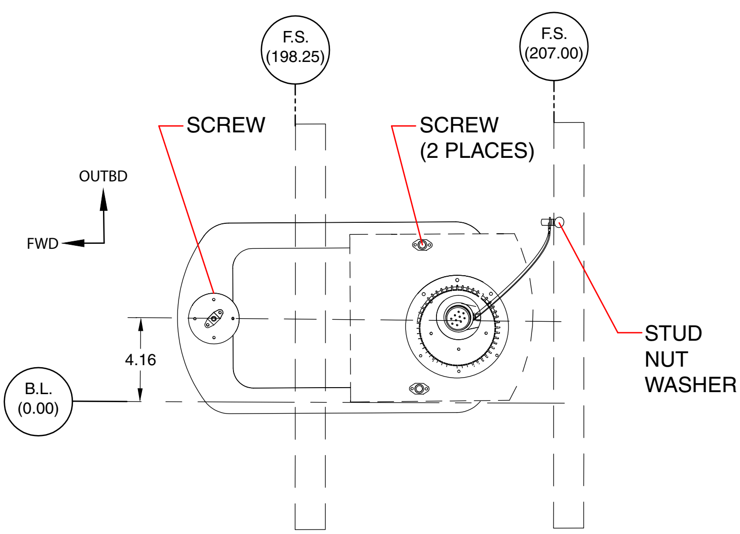
RA-3502 ADF Installation
Figure 3 (Sheet 1 of 3)



DETAIL A
ADF ANTENNA INSTALLATION
RHS LOOKING OUTBD



VIEW B - B
LOOKING AFT



VIEW C - C
TOP LOOKING DOWN

RA-3502 ADF Installation
Figure 3 (Sheet 2 of 3)

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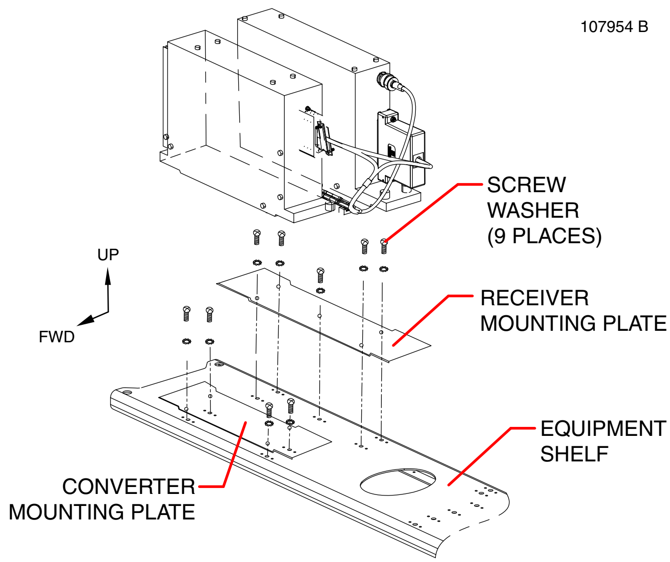
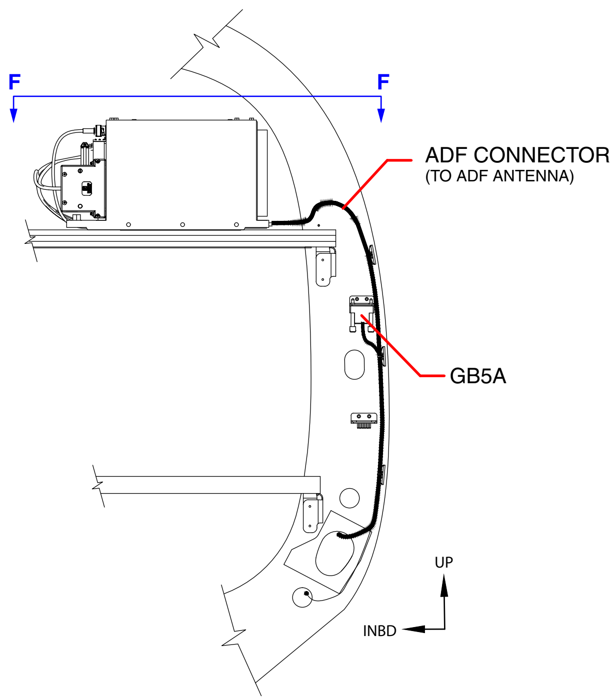
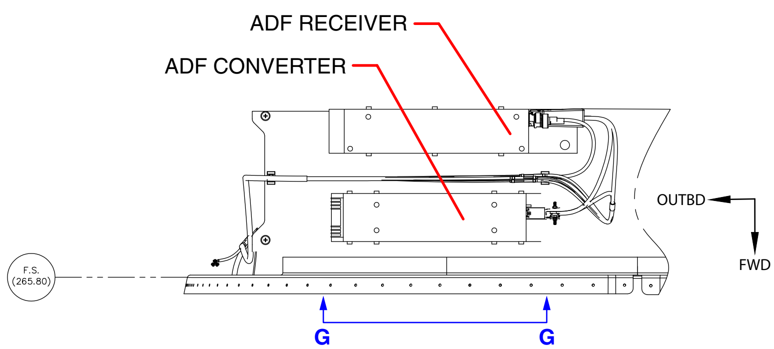


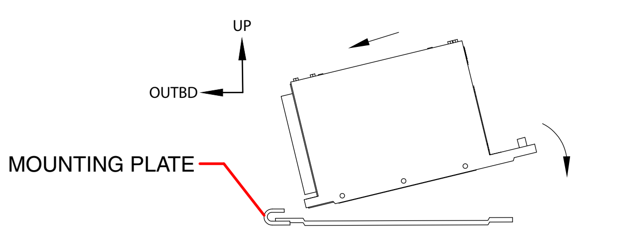
DETAIL D
ADF RECEIVER AND CONVERTER INSTALLATION
3D VIEW



VIEW E - E
RHS LOOKING FWD



VIEW F - F
TOP LOOKING DOWN



VIEW G - G
CONVERTER AND RECIEVER BRACKET MOUNTING
LOOKING FWD

RA-3502 ADF Installation
Figure 3 (Sheet 3 of 3)

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CHAPTER

35

OXYGEN

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CHAPTER 35

LIST OF EFFECTIVE PAGES

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CHAPTER 35 - OXYGEN

TABLE OF CONTENTS

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CREW

1. Fixed Oxygen Generator System (PA-46-350P)

   The oxygen generator system will deliver constant flow when activated for approximately 15 minutes (i. e. - sufficient oxygen supply for the airplane to be flown to an altitude of 10,000 feet or less).

   The pilot's and copilot's oxygen generators are located in a tray under the copilot's seat. A drawer containing two masks opens inboard. The basic system is the same as that installed for the passengers. See 35-20-00 for detailed information on removal, installation, and inspection.

2. Fixed Oxygen System (PA-46R-350T)

   The oxygen system will deliver constant flow when activated.

   The pilot's and copilot's oxygen masks are located in a storage pouch behind the copilot's seat, just aft of the wing spar box. A storage pouch containing two masks for the passengers is located just aft of the wing spar box, outboard of each forward passenger seat. See 35-20-00 for detailed information on removal, installation, and inspection.

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PASSENGER

1. Fixed Oxygen Generator System (PA-46-350P)

See "Figure 1" on page 35202 and "Figure 2" on page 35203.

The chemical oxygen generator system used in this airplane is a single-use system, which will deliver constant flow when activated for approximately 15 minutes (i.e. - sufficient oxygen supply for the airplane to be flown to an altitude of 10,000 feet or less) after ignition.

A. Description and Operation

Passenger oxygen is installed in the cabin under the passenger seat directly behind the copilot's seat, and contains four masks and two oxygen generators.

The generators are activated individually by removing a mask from the storage drawer and pulling the lanyard attached to the mask. This action pulls the firing pin allowing the hammer to strike the percussion cap and ignites the generator. Once activated, the oxygen will continue to flow until the particular generator is expended. An indicator light located in the annunciator panel (or amber CAS message located on MFD on Garmin G1000/G1000 NXi equipped aircraft) will illuminate whenever any one of the generators has been activated or expended.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Take care when handling the oxygen generators. Even though they are not under high pressure, they do generate very high temperatures. Inadvertent ignition due to careless handling is to be avoided. The generators have a percussion cap igniter on one end which must be protected with a plastic cover when the generator(s) are removed from the mounting tray/drawer. Do not remove this plastic protective cover until the generator is installed/re-installed in the mounting tray/drawer and the system is re-armed.

NOTE: Installed generators are normally removed only when expended, damaged, or at the end of their ten (10) year service life (as indicated by the date of manufacture on the individual unit dataplate).

B. 100 Hour Inspection

(1) Turn Battery master switch ON. Inspect the annunciator panel display to ensure "Oxygen" caution light remains OFF. On Garmin G1000/G1000 NXi equipped aircraft, ensure the "OXYGEN GEN OFF" CAS message is not displayed. If caution light is ON, or CAS message is displayed (Garmin G1000/G1000 NXi equipped aircraft) check first for an expended generator. If no generator is expended, check that the hammer has properly captured the microswitch at each generator location.

(2) Inspect each firing pin and hammer. If firing pin has been pulled and hammer is in contact with percussion cap, consider the generator to be expended and replace.

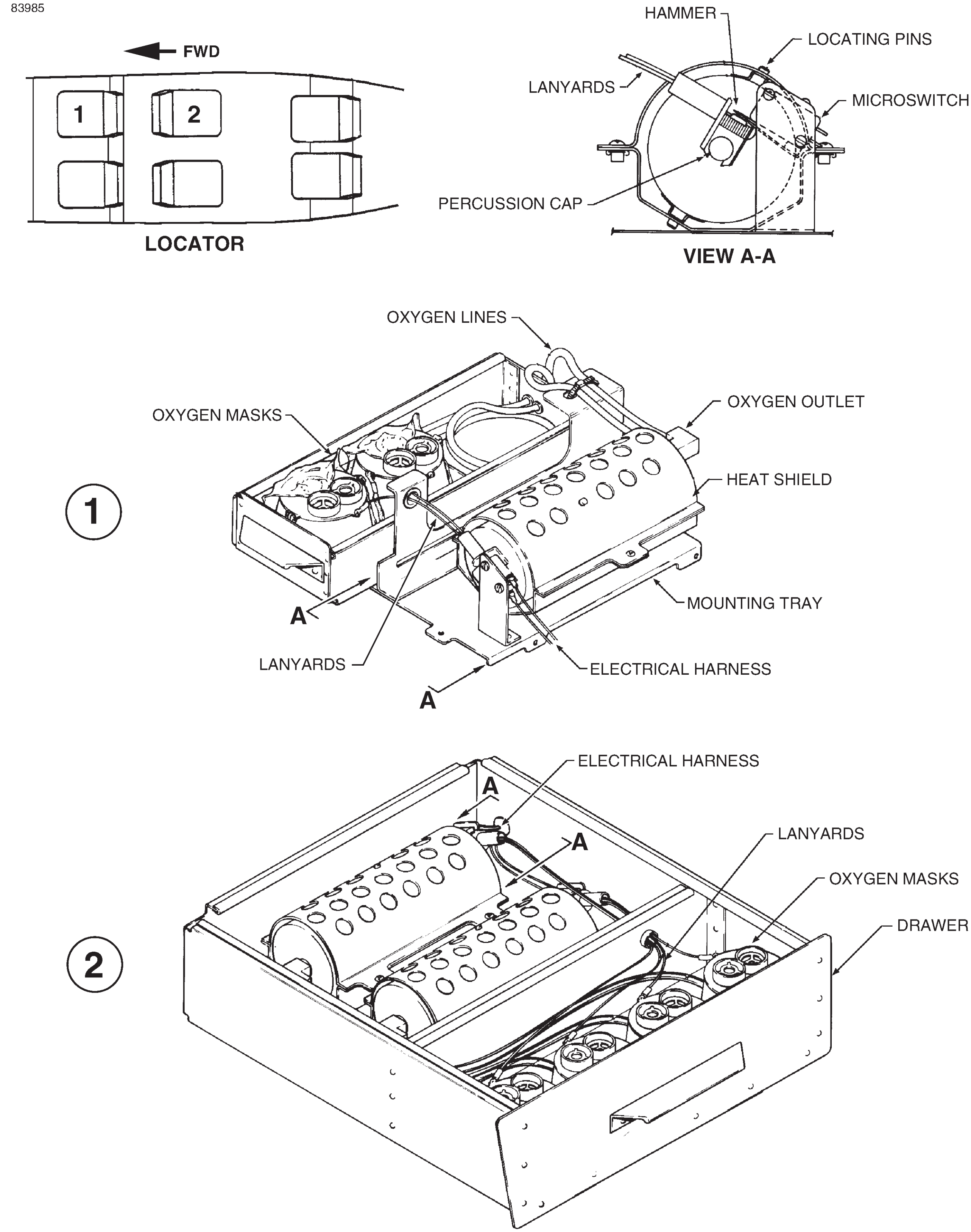
(3) Inspect each percussion cap:

(a) Verify that the plastic protective cover (see "Figure 2" on page 35203) has been removed from each percussion cap. If not, remove plastic protective cover.

(b) If the normally smooth surface of the percussion cap exhibits any evidence of indentation, replace the generator.

(4) Check each generator for security within its mounting points.

(5) Check each generator for any obvious physical damage (i.e. - punctures, dents, deep scratches, etc.). If damage present, replace generator.



Oxygen Generator System Installation
Figure 1

CAUTION: DO NOT PULL ON THE LANYARD. THE GENERATOR MAY IGNITE IF THE LANYARD IS PULLED.

(6) Check lanyard cords for fraying and security at both ends.

(7) Check oxygen lines for security of connections to mask and generator outlet fitting. Verify that each mask lanyard and oxygen line are connected to the same generator.

(8) Check the manufacturer's data plate on each oxygen generator for date of manufacture. If date is ten (10) years or older, replace generator.

C. Removal

WARNING: WHEN ACTIVATED, THE EXTERIOR SURFACE OF THE GENERATOR WILL REACH VERY HIGH TEMPERATURES (UP TO 500° F) UNTIL DEPLETION. IF REMOVING AN UNEXPENDED GENERATOR, TAKE CARE TO AVOID INADVERTENT ACTIVATION.

(1) Disconnect the wiring harness and remove the mounting tray/drawer from the aircraft.

(2) Remove the screws (3) securing the heat shield over the generator.

(3) Disconnect the oxygen lines from the generator outlet fitting.

(4) Untie the lanyards from the masks. Retain the sleeves installed over the knots for re-use.

(5) If generator has not been expended, place a plastic protective cover over the percussion cap and coil and secure the lanyard to ensure the firing pin is not pulled during removal and handling.

(6) Remove generator from mounting tray/drawer.

D. Installation

NOTE: New generators are delivered from the factory with the lanyards and firing pin installed and the hammer armed. A plastic protective cover protects the percussion cap from being ignited if the firing pin is inadvertently pulled during handling.

If the firing pin has been pulled during handling, verify that the generator has not been expended and then re-arm the generator by pulling the hammer back and inserting the firing pin attached to the end of the lanyard in front of the hammer. Ensure the firing pin covers at least 2/3 of the hammer.

(1) Place a new generator into position in the mounting tray/drawer, ensuring proper indexing of locator pins to the mounting brackets - i. e. - two down, one up.

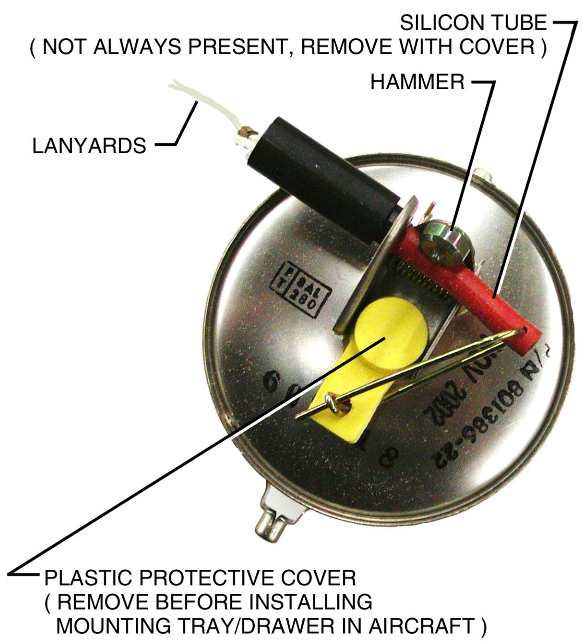
(2) Ensure that the hammer is in contact with the micro switch.

(3) Connect the oxygen lines to the generator outlet fitting.

(4) Place the heat shield over the generator. Verify insertion of the indexing pin in the heat shield and secure with screws (3).

CAUTION: ENSURE THAT THE LANYARD AND TUBE FROM EACH MASK ARE CONNECTED TO THE SAME GENERATOR.

(5) Trim lanyard lengths to 30 inches ± 1. Using a square knot, tie lanyards to the appropriate masks and cover knots with sleeves.



New Generator Fresh from the Box
Figure 2

(6) Coil up each mask and related oxygen line and lanyard and install in the storage compartment of the mounting tray/drawer.

CAUTION: FAILURE TO REMOVE THE PLASTIC PROTECTIVE COVER FROM THE PERCUSSION CAP WILL PREVENT THE GENERATOR FROM IGNITING WHEN THE LANYARD IS PULLED.

(7) Remove the plastic protective cover (see “Figure 2” on page 35203) from the percussion cap on the newly installed generator.

(8) Install mounting tray/drawer in aircraft.

(9) Connect the wiring harness, turn ON the Battery master switch, and verify the annunciator panel caution light remains out, or CAS message is not displayed (Garmin G1000/G1000 NXi equipped aircraft). If not recheck that the hammer has properly captured the microswitch.

2. Fixed Oxygen System (PA-46R-350T)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: WHEN REFILLING ANY OXYGEN CYLINDER MAKE SURE TO USE ONLY AVIATION BREATHING OXYGEN AS SPECIFIED IN MIL-PRF-27210G. THE MOISTURE CONTENT OF AVIATION OXYGEN CANNOT EXCEED 0.005 MILLIGRAMS OF WATER VAPOR PER LITER OF GAS AT 70°F (21°C) AND 29.92 INCHES OF MERCURY (760 MM HG.).

The following provides supplemental information for the servicing of the oxygen system. Major repairs to the oxygen system should be accomplished by an approved shop.

A. Description and Operation

See “Figure 3” on page 35206.

WARNING: DO NOT USE GREASE OR ANY TYPE OF GREASE FITTING ON ANY OXYGEN SYSTEM. WHEN WORKING WITH AN OXYGEN SYSTEM MAKE SURE HANDS, CLOTHING, TOOLS, AND IMMEDIATE AREA ARE FREE OF GREASE.

A fixed oxygen system is available in Matrix S/N’s 4692001 & up airplanes only, as standard equipment. The major components for this system are manufactured by AEROX Aviation Oxygen Systems. Accordingly, AEROX, as well as Piper Dealer Service Administrators (DSA’s), should be contacted for information / procedures not covered herein.

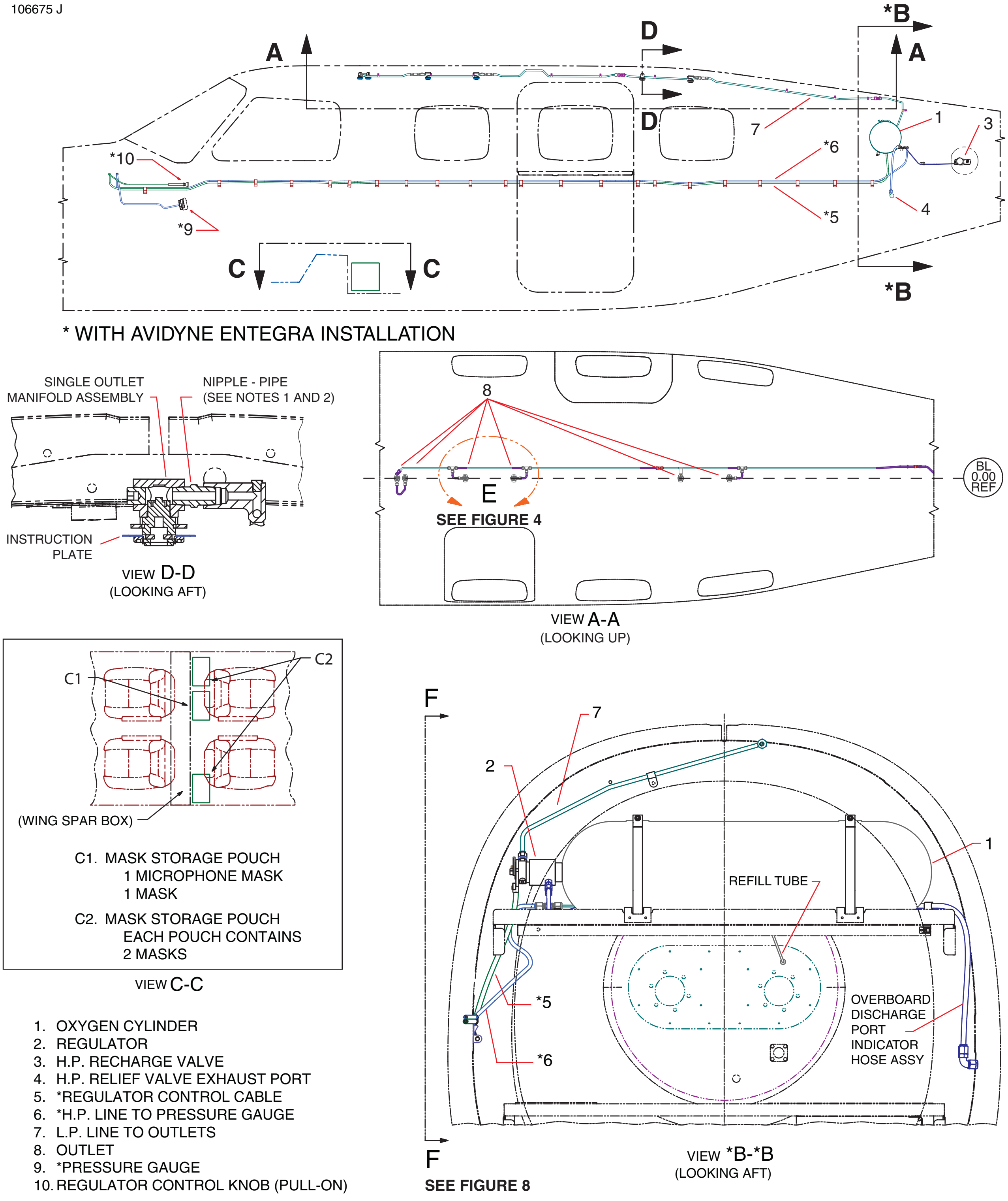
The oxygen system cylinder is installed in the tailcone, behind the baggage compartment, and is connected to an external recharge valve mounted behind a door on the left side of the fuselage, aft of FS 273.75. An overboard discharge port indicator is viewable on the left side of the fuselage at FS 270.91. The low pressure (L.P.) feed line for the outlets is routed from the right side of the fuselage and then up into the center of the cabin overhead. It joins the outlets’ distribution manifold at a union-fitting aft of the rear passenger outlet.

The PULL-ON regulator valve control knob and pressure gauge (in Avidyne Entegra-equipped aircraft) are installed in the pilot’s instrument panel. The PULL-On control knob is to the left of the standby instruments. The pressure gauge is to the left of the flap control. The control knob actuates a cable routed along the right side of the fuselage and attached to the oxygen system cylinder regulator valve. In Avidyne Entegra-equipped aircraft, the pressure gauge is fed by a high pressure line routed from the oxygen system cylinder along the right side of the fuselage where it joins the control cable and the two are then routed into the back of the instrument panel. In G1000 equipped aircraft, cylinder pressure is displayed on the MFD via a transducer.

NOTE: Oxygen cylinders are identified by the ICC or DOT identification stamped on the cylinder. The lightweight cylinder of composite construction (DOT-SP-10945) installed in these airplanes must be hydrostatically tested every 5 years, and the service life may not exceed 15 years. The month and year of the last test is stamped beneath the ICC/DOT identification.

B. Troubleshooting

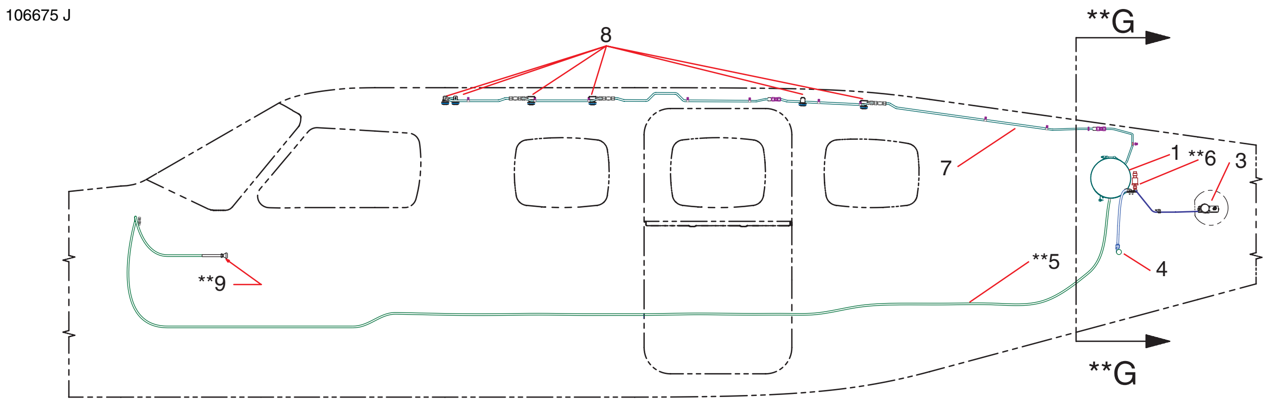
See “Chart 1”.



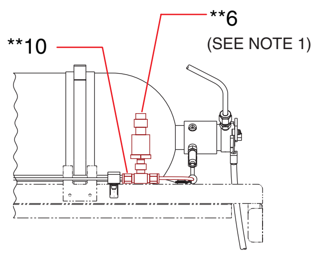
1. OXYGEN CYLINDER
2. REGULATOR
3. H.P. RECHARGE VALVE
4. H.P. RELIEF VALVE EXHAUST PORT
5. *REGULATOR CONTROL CABLE
6. *H.P. LINE TO PRESSURE GAUGE
7. L.P. LINE TO OUTLETS
8. OUTLET
9. *PRESSURE GAUGE
10. REGULATOR CONTROL KNOB (PULL-ON)

NOTE: (1) APPLY TEFLON TAPE TO ALL MALE PIPE THREADS.
(2) FITTINGS SHOULD BE TURNED IN BY HAND 2-1/2 ± 1 TURN. WRENCH TIGHTEN FITTING 1-3/4 TURNS AND CONTINUE TIGHTENING UP TO 1 ADDITIONAL TURN AS REQUIRED TO ACHIEVE DESIRED ALIGNMENT OF MATING PARTS. THE TOTAL NUMBER OF TURNS (BY HAND AND WRENCH) IS NOT TO EXCEED 6-1/4 TURNS.

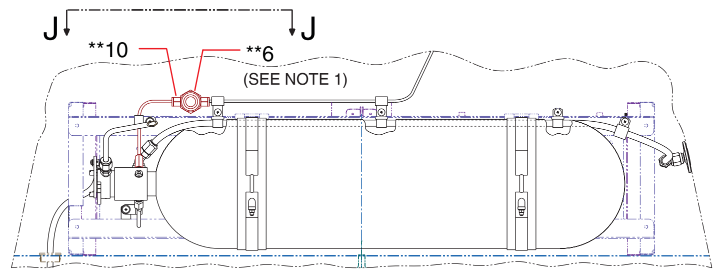
Fixed Oxygen System Installation
Figure 3 (Sheet 1 of 2)



**WITH GARMIN G1000 INSTALLATION

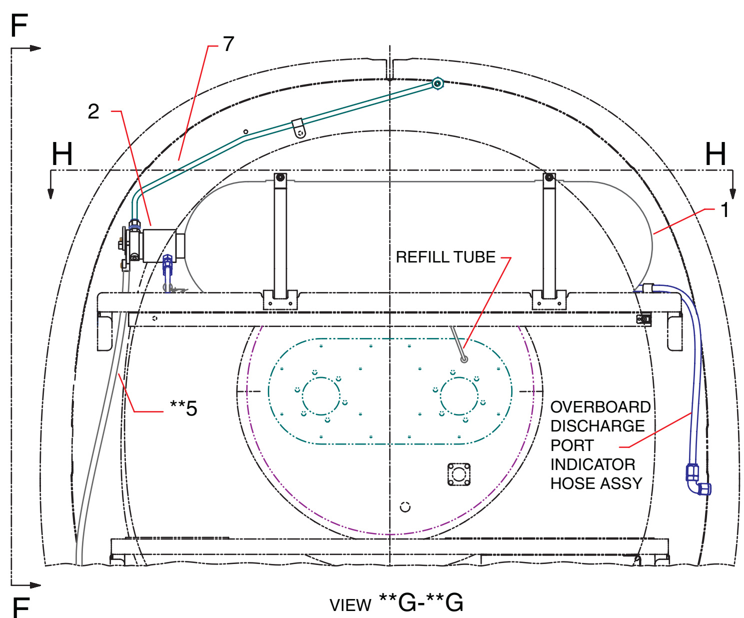


VIEW J-J
(LOOKING FORWARD)



VIEW H-H
(LOOKING DOWN)

1. OXYGEN CYLINDER
2. REGULATOR
3. H.P. RECHARGE VALVE
4. H.P. RELIEF VALVE EXHAUST PORT
5. **REGULATOR CONTROL CABLE
6. **TRANSDUCER TO MFD
7. L.P. LINE TO OUTLETS
8. OUTLET
9. **REGULATOR CONTROL KNOB (PULL-ON)
10. **BRANCH TEE



VIEW **G-**G
(LOOKING AFT)

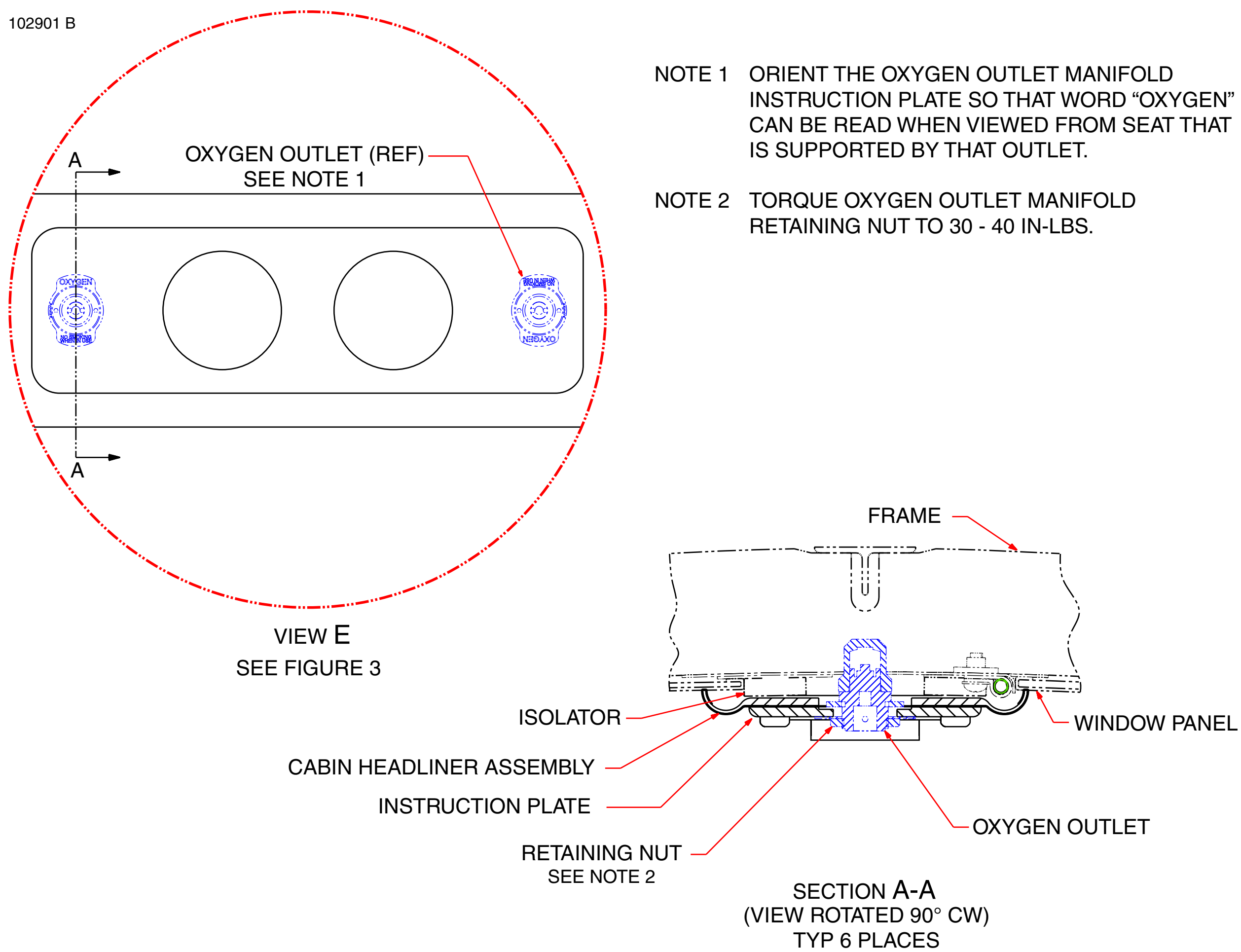
SEE FIGURE 8

NOTE: (1) APPLY TEFLON TAPE TO ALL MALE PIPE THREADS.
(2) FITTINGS SHOULD BE TURNED IN BY HAND 2-1/2 ± 1 TURN. WRENCH TIGHTEN FITTING 1-3/4 TURNS AND CONTINUE TIGHTENING UP TO 1 ADDITIONAL TURN AS REQUIRED TO ACHIEVE DESIRED ALIGNMENT OF MATING PARTS. THE TOTAL NUMBER OF TURNS (BY HAND AND WRENCH) IS NOT TO EXCEED 6-1/4 TURNS.

Fixed Oxygen System Installation
Figure 3 (Sheet 2 of 2)

CHART 1
TROUBLESHOOTING OXYGEN SYSTEM

Trouble	Cause	Remedy
No indication of pressure on pressure gauge.	Cylinder empty or leak in system has exhausted pressure.	Purge, charge, and check system for leaks.
	Pressure gauge or regulator defective.	Replace gauge.
Pressure indication normal but no oxygen flowing.	Oxygen cylinder regulator assembly defective.	Remove tank and have regulator removed.
Offensive odors in oxygen.	Cylinder pressure below 50 psi.	Purge the oxygen system.
	Foreign matter has entered the system during previous servicing.	Purge the oxygen system.



Headliner Overhead Outlet
Figure 4

C. Precautions

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

CAUTION: DO NOT ATTEMPT TO TIGHTEN ANY CONNECTIONS WHILE THE SYSTEM IS CHARGED.

CAUTION: BOTTLES WHICH HAVE BEEN EVACUATED TO 200 PSI FOR A SIGNIFICANT LENGTH OF TIME, OR THOSE THAT DO NOT PRODUCE AN AUDIBLE HISSING SOUND WHEN THE VALVE IS CRACKED, SHOULD BE REMOVED AND/OR PURGED. IF EITHER OF THESE CONDITIONS HAS EXISTED FOR A SIGNIFICANT LENGTH OF TIME, HYDROSTATICALLY TEST THE CYLINDER.

CAUTION: MAKE SURE THERE IS NO OIL, GREASE, HYDRAULIC FLUID, OR FUEL IN THE VICINITY OF ANY FITTINGS BEING SERVICED.

CAUTION: DO NOT USE THREAD LUBRICANTS OF ANY KIND. USE TEFLON TAPE (3M NO. 547) ON TAPERED PIPE THREADS, WITHOUT TAPE EXTENDING BEYOND THE FIRST THREAD. SEE APPLICATION OF TEFLON TAPE THREAD SEALANT, BELOW.

CAUTION: BEFORE WORKING WITH THE SYSTEM, MAKE SURE AIRCRAFT IS ELECTRICALLY GROUNDED AND YOUR HANDS TOOLS, AND CLOTHES ARE FREE OF OIL, GREASE AND DIRT.

D. Inspection and Maintenance

(PIR-PPS60087, Rev. U.)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Due to the nature of the process used to test compressed gas tanks, servicing and hydrostatic tests must be conducted by a DOT or manufacturer (AEROX) approved shop. The following material gives recommended inspection and maintenance information for the various parts of the oxygen system.

NOTE: Oxygen cylinders are identified by the ICC or DOT identification stamped on the cylinder. The lightweight cylinder of composite construction (DOT-SP-10945) installed in these airplanes must be hydrostatically tested every 5 years, and the service life may not exceed 15 years. The month and year of the last test is stamped beneath the ICC/DOT identification.

(1) Check the outlets for leakage both in the use and non-use condition and for leakage around an inserted connector. See Leak Test, below.

(2) Check the high pressure gauge for accuracy by comparing its indicated pressure with that of a gauge of known accuracy connected to the fill port.

(3) Visually inspect tank weekly. Look for dents, bulges, corrosion, major strap chaffing marks, and current hydrostatic test date. Should any of these problems exist, the tank should be removed and hydrostatically tested.

(4) System Flow Check

WARNING: THE FOLLOWING FLOW CHECK ALLOWS EXCESS OXYGEN TO FLOW INTO THE AIRFRAME. AVOID THE POTENTIAL FOR SPARK IGNITION BY NOT OPERATING ANY ELECTRICAL EQUIPMENT DURING THE TEST AND OBSERVE ALL SAFETY PRECAUTIONS RELATED TO OXYGEN HANDLING.

An operational check of the regulator can be accomplished as follows: (Refer to “Figure 5”.)

(a) Using an 18 inch (45.72cm) long hose having a 1/4 in. (0.635cm) I.D. x 1/2 in. (1.27cm) O.D., attach an AEROX 4110-626 FMV-PB1 flowmeter with plug-in to a test flow gauge. Connect the apparatus to the pilot’s outlet in the overhead panel.

(b) Insert an AEROX flowmeter in each of the other outlets.

(c) Pull the oxygen control knob to the ON position.

(d) Adjust all AEROX flowmeters by holding vertically and turning the black knob until the black indicator ball centers on the 25 thousand foot line on the “MASK AND STD CANNULA” scale. Do not use the “OXYSAVER” scale. The flow rate at sea level should be 3.7 to 5.1 liters per minute respectively.

(e) Repeat flow check at each position by interchanging the test flow gauge with individual flowmeters. Ensure all AEROX flowmeters remain set to 25 thousand feet, as noted above.

(f) Push the oxygen control knob to the OFF position and remove test flowmeter and all Aerox flow test tools.

(5) Check airframe logbook for last maintenance on oxygen system and perform as required per “Chart 2”.

(6) Test the oxygen for odor. Pure oxygen is odorless and tasteless. Any system having a significant odor present in the gas should be purged and the bottle replaced or removed and purged.

(7) Any fittings, connectors, and tubes which have imperfect threads, pitted or disfigured cones, or other damage should be replaced.

CAUTION: OXYGEN TUBES MUST NOT BE CLAMPED TO, OR SUPPORTED BY ELECTRICAL WIRE BUNDLES, HYDRAULIC OR OTHER LINES.

(8) Check plumbing for kinking, cracks, gouges, dents, deep scratches, or other damage. Replace as necessary.

(9) Oxygen Lines Clearance Check (Refer to “Figure 6”.)

Make sure to check the oxygen lines for proper clearance as follows:

(a) Two inch minimum between oxygen tubes and all flexible moving parts of the aircraft (flexible control cables, etc.). If enough space cannot be attained, protection from abrasion must be provided.

(b) At least 1/2 inch minimum between oxygen tubes and all rigid moving parts of the aircraft such as levers and rigid control rods.

(c) Six inch minimum separation between oxygen tubes and hydraulic, fuel and electrical system lines and components.

NOTE: When the six inch requirement cannot be complied with, one inch is allowed as long as electrical cables and other lines are supported at least every two inches; and, the oxygen tube(s) is protected by rubber neoprene hose fastened in place with cable ties at the location the specific item crosses or is near the oxygen tube(s) as shown in Figure 6, Sketch C. If an item is near the oxygen tube for a certain distance the oxygen tube for that distance must be covered.

(d) A minimum of 1/8 inch between tubing and structure adjoining the supporting clamp as shown in “Figure 6”, Sketch A.

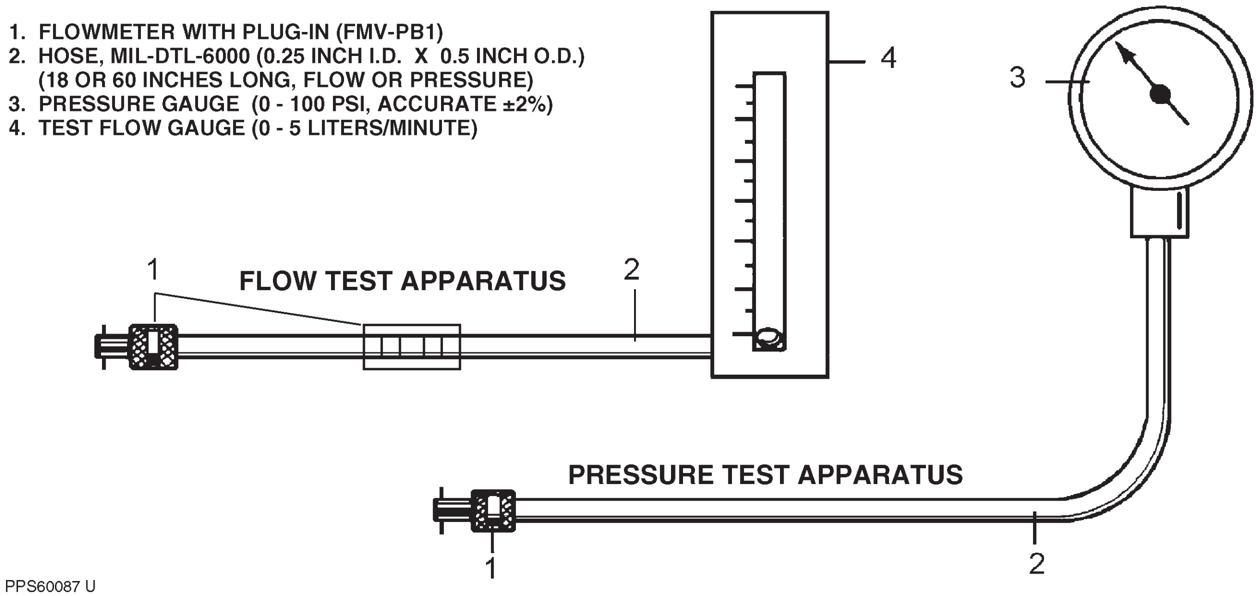
(e) Where a tube passes through a grommet, the tube must not bear on the grommet in any way that might cause cutting of the grommet in service as shown in “Figure 6”, Sketch D.

(f) While in service, items may receive vibrations causing them to come in contact with other parts of the aircraft. With this in mind, low pressure tubing that is supported well enough to prevent relative motion must have at least a minimum clearance of 1/8 inch from a projection (bolt, nut, etc). Low pressure tubing that cannot be supported well enough to prevent motion must have a minimum clearance of 1/8 inch allowed after the maximum travel of the tube. High pressure lines are affected similarly but require 1/2 inch minimum clearances. (Refer to “Figure 6”, Sketch B.)

(10) Perform any other required maintenance as directed in AC43.13-1 latest revision, Chapter 8.

(11) Clean components as necessary per Oxygen System Components, Cleaning and Purging, below.

1. FLOWMETER WITH PLUG-IN (FMV-PB1)
2. HOSE, MIL-DTL-6000 (0.25 INCH I.D. X 0.5 INCH O.D.) (18 OR 60 INCHES LONG, FLOW OR PRESSURE)
3. PRESSURE GAUGE (0 - 100 PSI, ACCURATE ±2%)
4. TEST FLOW GAUGE (0 - 5 LITERS/MINUTE)



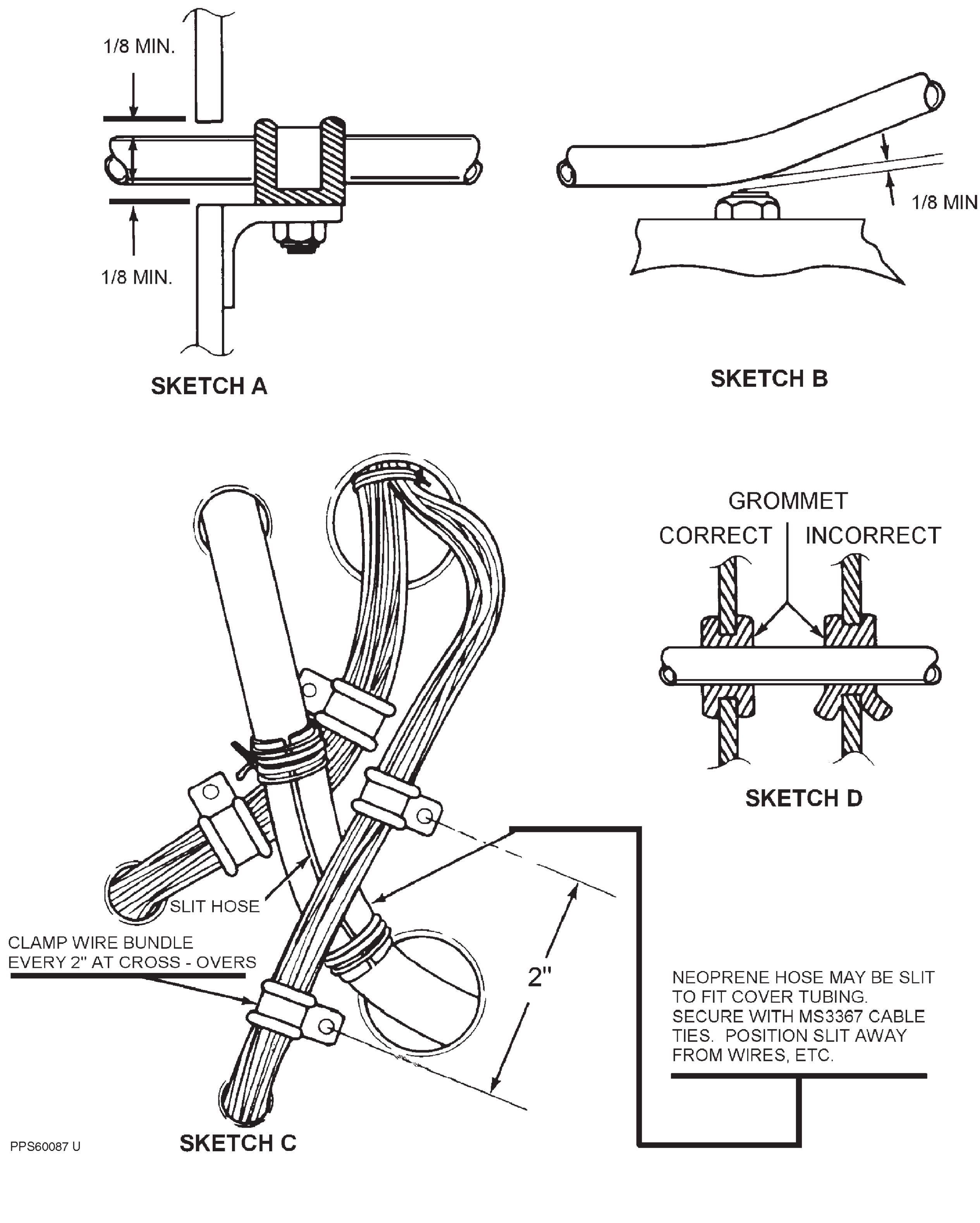
Oxygen System Test Apparatus
Figure 5

CHART 2
FIXED OXYGEN SYSTEM COMPONENT LIMITS

Component	Inspection	Overhaul
Cylinder	Every 90 Days [1]	Each 5 Years
Regulator	On Condition / Each Use [2]	Each 5 Years
Pressure Gauge	On Condition / Each Use [2]	Replace On Condition
High Pressure Lines	On Condition / Each Use [2]	Replace On Condition
Low Pressure Lines	On Condition / Each Use [2]	Replace On Condition
Outlets	On Condition / Each Use [2]	Replace On Condition
External Recharge Valve	On Condition / Each Use [2]	Replace On Condition [3]
Masks	On Condition / Each Use [2]	Each 3 Years

NOTES:

1. Visual inspection for dents, bulges, corrosion, or chafing.
2. Visual inspection in the normal course of use.
3. If the screen in front of valve is dirty, replace valve. Valve replacement is recommended every 5 years.



Oxygen Tubing Installation
Figure 6

E. Swagelok Fitting Installation

Refer to “Figure 7”.

NOTE: The high pressure line fitting at the regulator should be tightened until it bottoms. Make sure to use teflon tape on all male pipe threads.

(1) For Swagelok fittings not preswaged or for in-aircraft installation, proceed as follows:

(a) Turn the fitting nut onto the fitting finger tight and insert the tube until it bottoms firmly on the shoulder in the fitting.

(b) Tighten the nut with a wrench until the tube will not turn by hand.

(c) Mark the nut at the six o’clock position.

(d) Hold the fitting body steady with a backup wrench and tighten as follows:

1) On tubing with a diameter bigger than 3/16 inch, tighten 1 1/4 turns (to the nine o’clock position).

2) On tubing of 1/16, 1/8, and 3/16 inch diameter, tighten only 3/4 turn.

(e) If nut and tube must be disconnected from the fitting, reconnect by seating the tube on the shoulder of the fitting and tightening the nut finger tight. Follow up by tightening the nut with a wrench, one quarter turn (if absolutely necessary the original 1 1/4 or 3/4 tight position) and then snug with wrench.

(2) Preswaged Swagelok fittings are fabricated and installed as follows:

(a) Assemble the nut and ferrules finger tight on the preswaging tool and insert the tube until it firmly bottoms on the shoulder in the tool. The preswaging tool can be attained from Crawford Fitting Company, refer to List of Consumable Materials in Chapter 91.

(b) Tighten the nut on the fitting just enough that the tube within the fitting will not turn by hand.

(c) With a wrench, tighten the nut as follows:

1) On tubing with diameters over 3/16 inch, tighten 1 1/4 turns.

2) On tubing with 1/16, 1/8, or 3/16 inch diameter, tighten 3/4 of a turn.

(d) Unscrew the nut to release the ferrule-tube assembly from the tool.

(e) The assembly is installed on the fitting as follows:

1) Slide tube in fitting until it bottoms, turn nut to finger tight position and tighten one quarter turn with wrench.

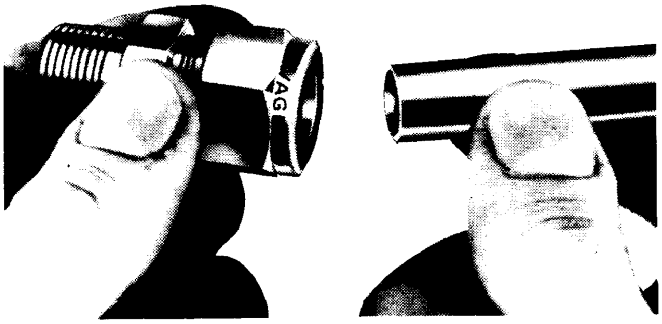
2) Snug slightly with wrench.

F. Teflon Tape Thread Sealant

All male pipe (tapered) threads of the oxygen system should be sealed with 3M No. 547 teflon tape. Teflon tape should not be used on straight threads. Do not use any other lubricants in place of the teflon or on any other threads.

(1) Wrap tape on the threads, starting with those farthest from the opening, in the direction of the thread spiral. Circle the threads, making sure that each side of the tape has a slight overlap.

(2) Wrap the tape such that it does not extend beyond the last thread on the fitting at the opening. The tape should then be pulled till it separates. Do not cut tape, it will not stick properly.

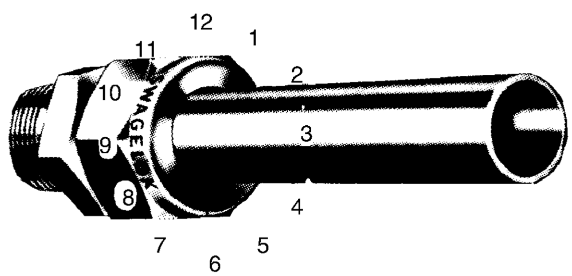


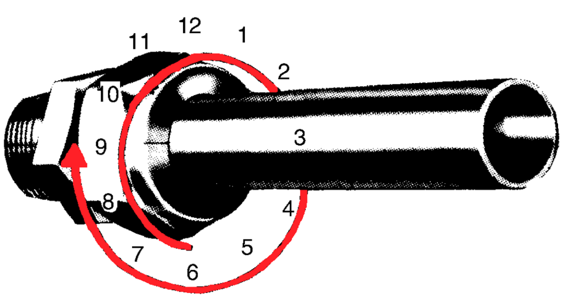
STEP 1

TURN THE FITTING NUT ONTO THE FITTING FINGER TIGHT AND INSERT THE TUBE UNTILL IT BOTTOMS FIRMLY ON THE SHOULDER IN THE FITTING

STEP 2

MARK THE NUT AT THE SIX O' CLOCK POSITION





STEP 3

HOLD THE FITTING WITH A WRENCH AND TIGHTEN THE FITTING NUT AS FOLLOWS:

A. TUBING WITH A DIAMETER GREATER THAN 3/16 INCH SHALL BE TIGHTENED 1 - 1/4 TURNS (THE NINE O' CLOCK POSITION)

B. TUBING WITH A DIAMETER OF 1/16, 1/8, OR 3/16 INCH SHALL BE TIGHTENED ONLY 3/4 TURN.

Swagelok Fitting Installation Procedure
Figure 7

G. Leak Test

Solution recommended for leak testing is Leak-Tec Formula #16-OX. Refer to the List of Consumable Materials for consumer information.

(1) Remove the cabin rear closeout panel and, with oxygen system turned off, disconnect the low pressure supply line and connect it to a regulated cylinder charged with dry nitrogen.

NOTE: Whenever a leak check is performed, all fitting connections as well as other questionable areas, should be inspected.

(2) Apply the leak detector solution to the test surface and watch for indication of leakage.

(3) Large leaks will produce bubbles immediately, but small leaks will form a white foam in 5 to 60 seconds.

(4) With outlets vacated of masks, connect a test pressure gauge to the co-pilot's outlet, which is the second outlet from the front of the overhead panel. Using an 60 inch (152.4cm) long hose having a 1/4 in. (0.635cm) I.D. x 1/2 in. (1.27cm) O.D., attach an AEROX PB1 flowmeter with plug-in to a sensitive pressure gauge having a range of 0 to 100 psi. See also Figure 5.

(5) Adjust the regulator on the dry nitrogen cylinder for 100 psi and check for leakage at the outlets.

(6) Correct any leaks and wipe off excess leak detector solution.

(7) Close the valve on the nitrogen gas tank and insert a Aerox flowmeter to relieve system pressure until test gauge reads zero.

(8) Disconnect test gauge, Aerox flowmeter, and nitrogen tank.

(9) If the oxygen cylinder is not to be hooked up or installed immediately, cap and cover the exposed fittings with new clean plastic bags. Temporarily support lines as needed to prevent damage. Make sure caps and coverings are as clean as possible.

H. Oxygen System Components

Keeping in mind the effect of compressed oxygen on materials, oxygen system components must be handled carefully. Ports on regulators, indicators, and other opened components must also be kept capped or plugged to prevent ingestion of foreign material. Adjustments or modifications should only be initiated under the auspices of the FAA, Piper, or AEROX.

NOTE: Replacement time for the recharge valve is on condition, but recommended each 5 years. The regulator requires overhaul every six (6) years. The lightweight composite cylinder must be removed every five (5) years for hydrostatic testing.

(1) Cleaning and Purging

CAUTION: CARE MUST BE EXERCISED TO PREVENT CONTAMINATION OF COMPONENTS BY OIL, GREASE, WATER, OR FOREIGN MATTER. COMPRESSED AIR USED IN CLEANING AND FLUSHING TUBES MUST BE CLEAN, DRY, FILTERED (OIL FREE) AIR ONLY.

(a) Three methods are recommended for cleaning oxygen system components:

1) Method I

a) Vapor degrease part(s) with trichlorethylene.

b) Blow part(s) dry with a stream of compressed air or dry nitrogen. See CAUTION, above.

2) Method II

a) For tubing, flush with naptha per specification TT-N-95.

b) Blow clean and dry off all solvent with clean, dry, filtered air. See CAUTION, above.

c) Flush with isopropyl alcohol.

d) Rinse thoroughly with fresh water.

e) Dry with air as described in previous caution or by heating at a temperature of 250° to 300°F for one-half hour.

NOTE: Solvents can be reused provided they do not become badly contaminated with oil. This condition can be determined by thoroughly evaporating 100 millimeters of the liquid in a glass dish of a determined weight. Evaporation may be accomplished by heating the dish at 200°F (93°C) for one-half hour. If after evaporation and cool down, the residue exceeds 100 milligrams in weight, the solvent cannot be used for this purpose.

3) Method III

a) Flush with hot inhibited alkaline cleaner until free from oil and grease.

b) Rinse thoroughly with fresh water.

c) Dry thoroughly with a stream of clean air as described in the previous CAUTION or by heating 250°F to 300°F (121°C to 149°C) for one-half hour minimum.

CAUTION: DO NOT USE ADHESIVE TAPE FOR ATTACHING OR SECURING PROTECTIVE COVERINGS ON OXYGEN COMPONENTS. USE WAXED LACING TWINE OR TIE WRAPS.

(b) After cleaning, all tubing must be protected by caps, plugs and/or plastic bags.

(c) Before installation, make sure fitting, tube, and fixture threads are in good condition and that the cones do not exhibit pitting or disfigurement.

(2) Oxygen Cylinder

Refer to “Figure 8”.

NOTE: Procedures for the Maintenance and Overhaul of the Oxygen Cylinder Valve Assembly (P/N 766-584) are contained in the Aerox Component Maintenance Manual (CMM) for the 4110-121 Series, Manual No. 35-00-02. See Supplementary Publications in the Introduction.

(a) Removal

CAUTION: BEFORE ENTERING THE TAIL CONE, FIRST SUPPORT THE AIRCRAFT TAIL WITH A SUITABLE TAILSTAND.

From inside the aft cabin:

1) Remove screws attaching finished bulkhead to fuselage bulkhead.
2) Remove finished bulkhead.
3) With immediate area clear of flammables (grease, hydraulic fluid, fuel, etc.) and oxygen system off; connect a mask or tube to an outlet to exhaust any pressure in the system.

NOTE: Continuous pressure is applied to high pressure line until it is disconnected from cylinder. A check valve will close when high pressure line is disconnected from cylinder. The closing of this valve is frequently accompanied by a loud popping sound.

4) Carefully unscrew high pressure feed/recharge line at regulator until pressure decreases and then remove line. Cap line immediately after removal.
5) Disconnect high pressure relief line from regulator. Cap line immediately after removal.
6) Disconnect low pressure line from regulator. Cap line immediately after removal.
7) Loosen and open clamps securing oxygen cylinder to its shelf.
8) If necessary, move cylinder slightly to gain access to regulator valve control arm. Disconnect regulator valve control cable from cylinder / control arm by unscrewing the cable fitting set screw which secures the cable core to the regulator lever. Take care not to kink cable.

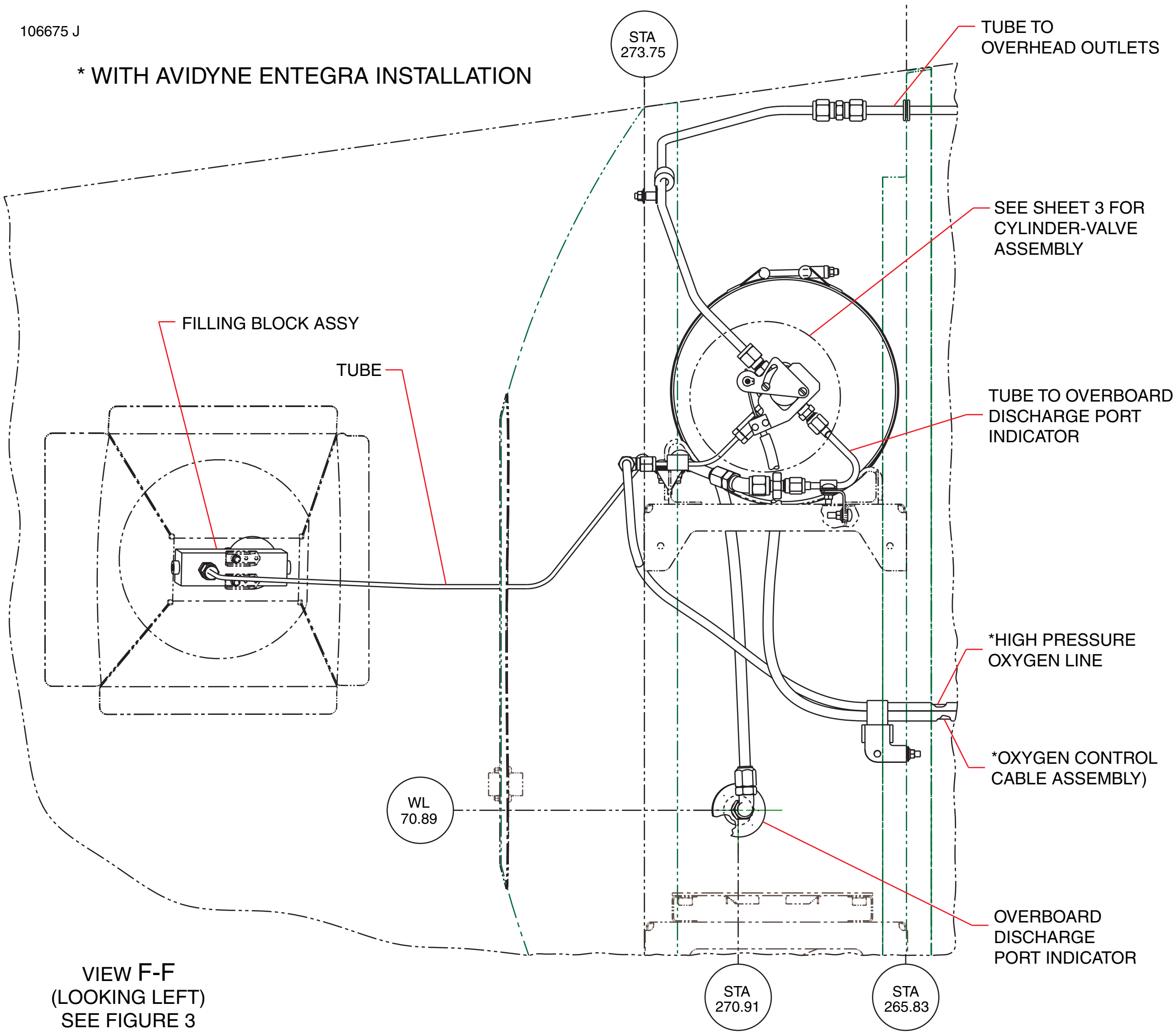
CAUTION: OPENING CONTROL VALVE DURING REMOVAL OF OXYGEN LOW PRESSURE LINE FROM CYLINDER WILL RESULT IN AN UNCHECKED FLOW OF OXYGEN INTO BAGGAGE COMPARTMENT UNTIL VALVE CAN BE CLOSED.

9) Safety valve on cylinder in the OFF position.
10) Remove cylinder from airplane.

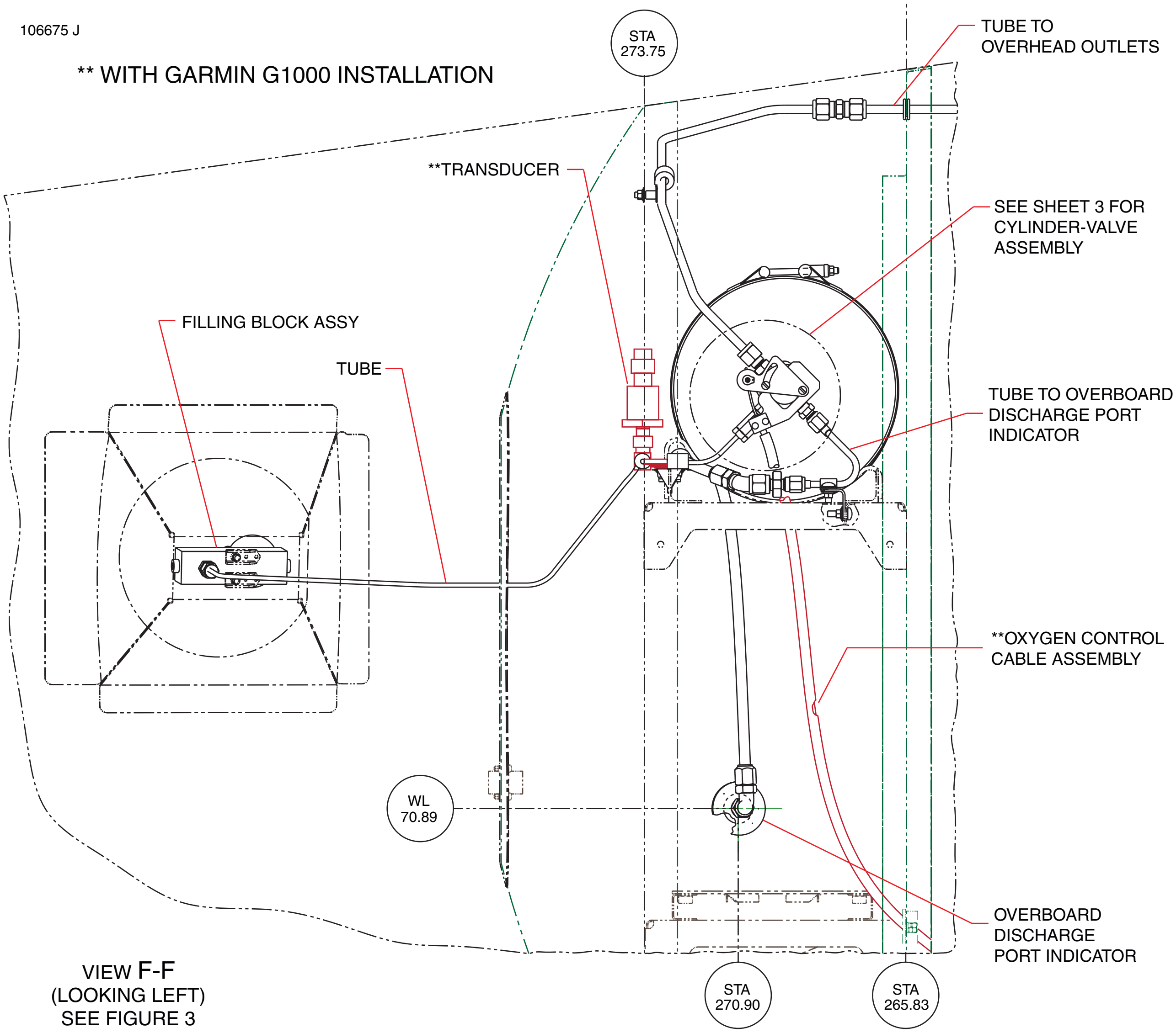
(b) Installation

1) If cylinder mounting shelf has been removed, reinstall it first.
2) Position cylinder in airplane as shown in “Figure 3” and “Figure 8”. Ensure that regulator valve control arm is free to move and does not contact surrounding area.
3) Attach and secure regulator valve control cable, before securing cylinder to shelf.
4) Install and secure two cylinder hold-down clamps.
5) Connect L. P. line to regulator. Insert tubing into fitting until ferrule seats in fitting. Tighten the nut by hand and then one quarter turn with a wrench. If fitting is relatively new the nut might be turned 3/4 of a turn. Follow up by snugging the nut slightly with a wrench.

NOTE: Apply teflon tape to all tapered male threads as advised in “Teflon Tape Thread Sealant” on page 352014.

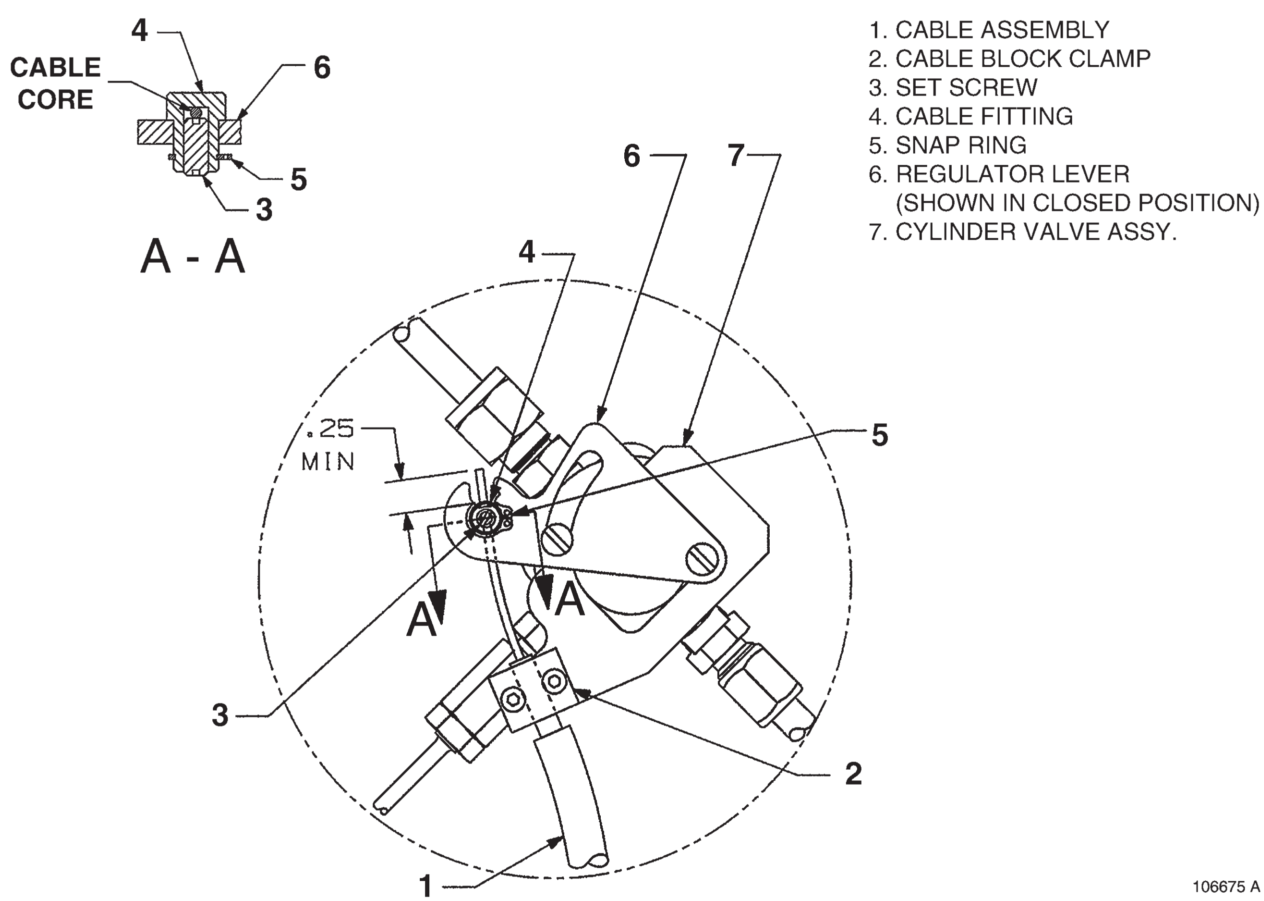


Oxygen Cylinder and Regulator Valve
Figure 8 (Sheet 1 of 3)



Oxygen Cylinder and Regulator Valve
Figure 8 (Sheet 2 of 3)

6) Connect H. P. feed/recharge and relief lines to regulator. Insert tubing into fitting until ferrule seats in fitting. Tighten the nut by hand and then one quarter turn with a wrench. If fitting is relatively new the nut might be turned 3/4 of a turn. Follow up by snugging the nut slightly with a wrench.
7) Unsafety valve on cylinder. Check that valve remains in OFF position.
8) Check pressure and refill bottle as necessary.
9) Inspect for leaks, especially at fittings that have been separated.
10) Reinstall finished bulkhead in aft cabin, secure with screws.
11) If used, remove tailstand from airplane.



Oxygen Cylinder and Regulator Valve
Figure 8 (Sheet 3 of 3)

(3) Pressure Gauge

Installed in S/Ns 4692001–4692133 with Avidyne Entegra. See “Figure 9”.

NOTE: In Garmin G1000-equipped PA-46R-350Ts, a transducer (see “Figure 3” on page 35206 and “Figure 8” on page 352019) provides oxygen pressure information to the MFD. See Oxygen Pressure Transducer, below.

The oxygen system flourescent panel pressure gauge assembly is installed in the bottom center of the instrument panel slightly left of the flap control handle. Access is obtained from beneath the instrument panel.

The pressure gauge is tied into the same high pressure (H.P.) line as the recharge valve, through T-fitting near the tank regulator-control valve.

(a) Removal

CAUTION: BEFORE ENTERING THE TAIL CONE, FIRST SUPPORT THE AIRCRAFT TAIL WITH A SUITABLE TAILSTAND.

1) At the oxygen cylinder:

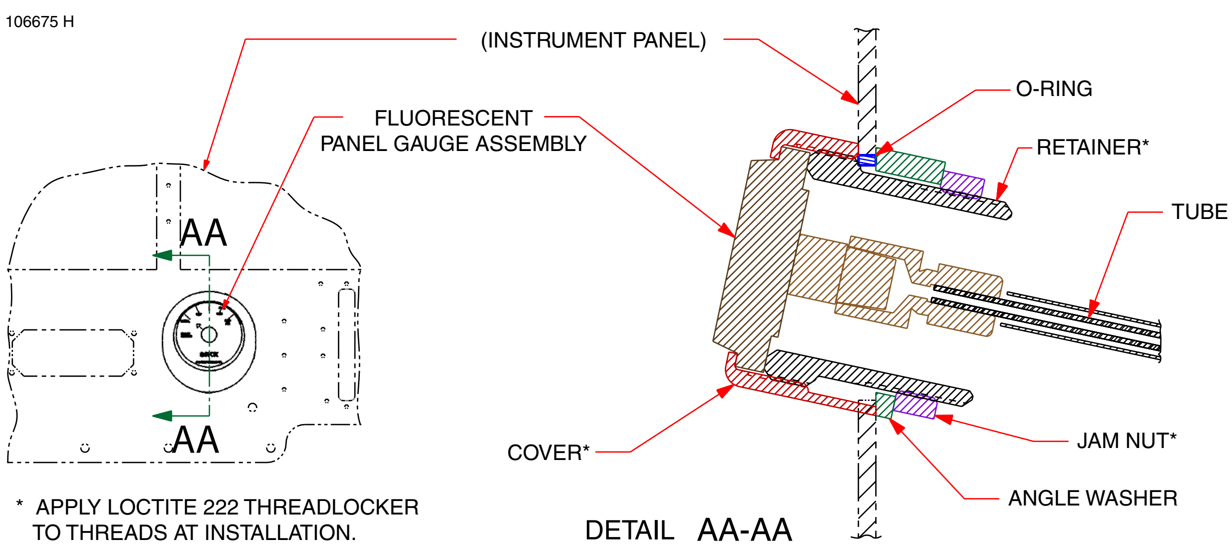
a) Remove screws attaching finished bulkhead to fuselage bulkhead.

b) Remove finished bulkhead.

c) With immediate area clear of flammables (grease, hydraulic fluid, fuel, etc.) and oxygen system off; connect a mask or tube to an outlet to exhaust any pressure in the system.

NOTE: Continuous pressure is applied to high pressure line until it is disconnected from cylinder. A check valve will close when high pressure line is disconnected from cylinder. The closing of this valve is frequently accompanied by a loud popping sound.

d) Carefully unscrew high pressure line until pressure decreases and then remove line. Cap line immediately after removal.



Effectivity
4692001–4692133, Avidyne

Pressure Gauge
Figure 9

2) At the instrument panel:

a) Disconnect high pressure line from gauge and cap immediately.

b) Remove jam nut and angle washer securing gauge retainer to panel.

c) Remove gauge from pilot's side of instrument panel.

(b) Installation

1) Apply two-sided tape (P/N 380-011) to back of gauge assembly and trim to fit.

2) Assemble gauge assembly into cover, ensure proper gauge alignment, and secure with retainer.

3) Install O-ring over retainer.

4) Install gauge/cover/retainer/O-ring assembly into instrument panel by inserting retainer through instrument panel and securing with angle washer and jam nut.

5) Verify correct readable alignment of the pressure gauge.

6) Connect H. P. line to gauge.

7) Connect H. P. line to cylinder.

8) Inspect fittings that have been separated for leaks.

9) Reinstall finished bulkhead in aft cabin, secure with screws.

10) Remove tailstand from airplane.

(4) Recharge Valve

See "Figure 10".

(a) Removal

The recharge valve is located on the left rear side of the aircraft and is covered by its own access door. This valve is connected to a T-fitting which interconnects the H. P. line from the regulator and the H. P. gauge feeder line. Accordingly, the recharge valve and its line are under constant cylinder pressure as long as the H. P. line is connected to the regulator. Access to the H. P. line connection is through the right rear fuselage access panel.

1) Remove the tailcone access panel.

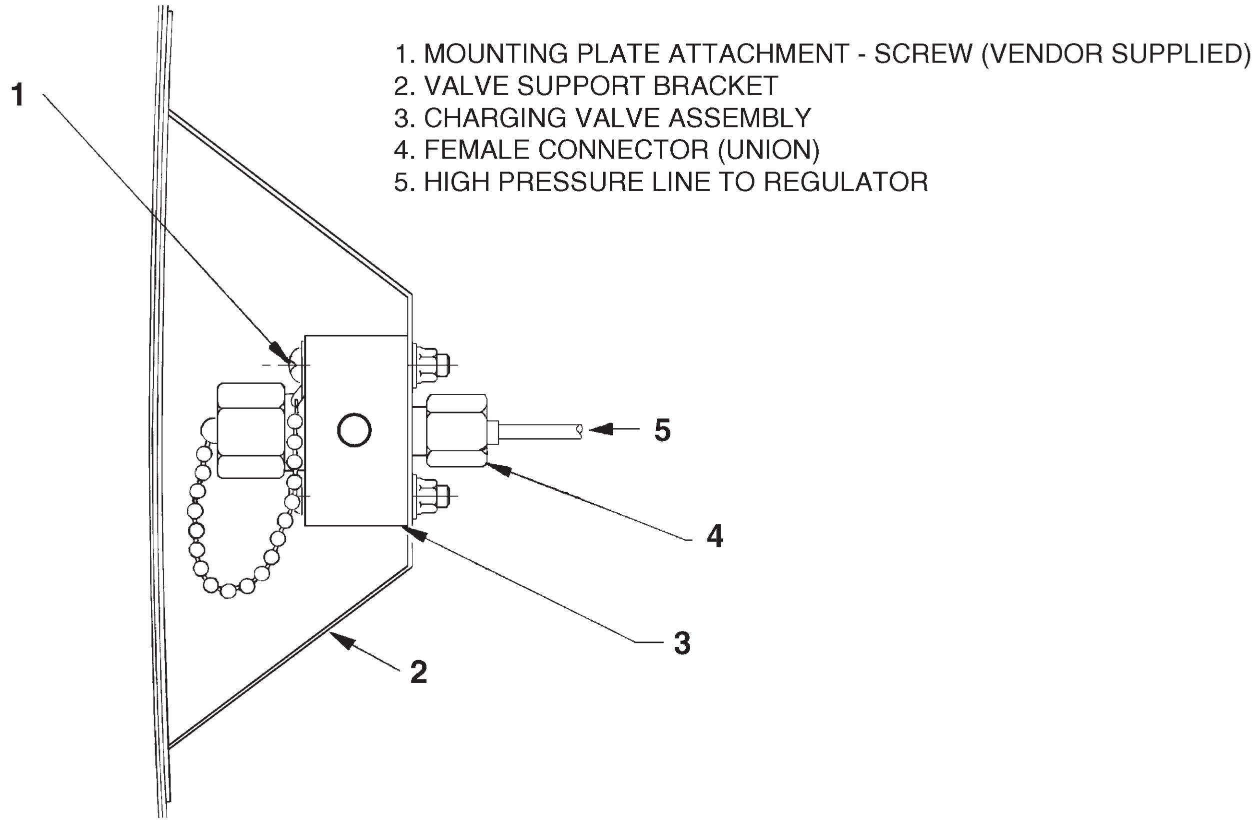
NOTE: Continuous pressure is applied to high pressure line until it is disconnected from cylinder. A check valve will close when high pressure line is disconnected from cylinder. The closing of this valve is frequently accompanied by a loud popping sound.

2) Disconnect the high pressure line from the recharge valve assembly. Cap line immediately after removal.

3) Access recharge valve by opening the door on the left of the fuselage.

4) Remove two screws securing the recharge valve to the bracket.

5) Remove valve from airplane.



Oxygen System Recharge Valve Installation
Figure 10

(b) Installation

NOTE: Apply teflon tape to all tapered male threads as advised in "Teflon Tape Thread Sealant" on page 352014.

1) Insert valve assembly onto bracket accessed through door on left aft fuselage.
2) Align screw holes in recharge valve with those in mounting bracket.
3) Install the two mounting screws. Attach cap chain, with information plate attached, with one of the screws.

CAUTION: CONNECT HIGH PRESSURE LINE TO VALVE BEFORE CONNECTING TO CYLINDER.

4) Connect H. P. line to recharge valve.
5) Reinstall the tailcone access panel.

(5) Outlets

Refer to “Figure 3” and “Figure 4”.

(a) Removal

1) Check that the oxygen system is completely turned off. Insert an oxygen mask to release pressure and ensure the system is off.

2) With a suitable spanner wrench, unscrew and remove the outlet retainer ring(s) and information plate(s).

3) Remove or drop overhead panel sufficiently to gain access to low pressure (L. P.) line connections.

4) Disconnect outlet(s) from L. P. feed lines. As appropriate: disconnect the T-unions or the elbow-union connecting the outlet to main L. P. feed line.

5) Remove outlet(s) from airplane.

(b) Installation

1) Position outlet(s) in airplane.

2) Connect outlet(s) to L. P. feed lines. As appropriate: connect the T-unions or the elbow-union connecting the outlet to main L. P. feed line.

3) Inspect fittings that have been separated for leaks.

4) Replace overhead paneling and secure in place.

5) For each outlet, position the information plate so that the word “OXYGEN” can be read when viewed from the seat that is supported by that outlet. With a suitable spanner wrench, screw on the outlet retainer ring.

(6) Oxygen On/Off Control

The oxygen system PULL ON (push off) control knob is located to the left of the standby instruments on the pilot’s instrument panel. Access is obtained from beneath the instrument panel.

(a) Removal

CAUTION: BEFORE ENTERING THE TAIL CONE, FIRST SUPPORT THE AIRCRAFT TAIL WITH A SUITABLE TAILSTAND.

1) Disconnect cable from regulator-control mechanism on cylinder:

a) Remove screws attaching finished bulkhead to fuselage bulkhead.

b) Remove finished bulkhead.

c) f necessary, move cylinder slightly to gain access to regulator valve control arm. Disconnect regulator valve control cable from cylinder / control arm by unscrewing the cable fitting set screw which secures the cable core to the regulator lever.

d) Release cable from all clamps and straps securing cable to H. P. Gauge feed line. Note position of straps for reinstallation.

2) Remove cabin right side panels sufficiently to gain access to the control cable running the entire length of the cabin.

3) Release cable from all clamps and straps securing cable to H. P. Gauge feed line, both in the cabin sidewall and under the instrument panel. Note position of straps for reinstallation.

4) Remove retaining nut from rear of control knob.

5) Pull cable from airplane through instrument panel. Retrieve retainer nut as cable end pulls through the instrument panel.

(b) Installation

1) Insert cable through instrument panel. Slide retainer nut onto cable and secure control knob to instrument panel.
2) Feed cable up through straps and clamps forward of the instrument panel to right side of the cabin where it joins the H. P. Gauge feed line to cylinder.
3) Secure cable to H. P. Gauge feeder line and structure with same number of straps and clamps installed at same locations as those loosened to remove cable.
4) Trim cable core to allow sufficient material to feed thru the cylinder-valve assembly cable block clamp and cable fitting. A minimum of .25 inch of cable core should protrude above the fitting. (Refer to Figure 8, Sheet 2 of 2.)
5) Secure cable core in cable fitting with set screw.
6) Secure cable with retaining screw and clamp block clamp to cylinder valve assembly.
7) Check operation, adjust as required.
8) Replace and secure sidewall panels in the cabin.
9) Replace and secure finished bulkhead in aft cabin. If used, remove tailstand.

(7) Oxygen Pressure Transducer

Installed in S/N's 4692134 and up with Garmin G1000 only.

See "Figure 3" on page 35206 and "Figure 8" on page 352019.

A pressure transducer replaces the oxygen pressure gauge in Garmin G1000 equipped PA-46R-350Ts, and provides pressure data to the G1000 MFD. The transducer is located near the oxygen cylinder in the aft fuselage, connected to the HP pressure line to the recharge valve.

(a) Removal

CAUTION: BEFORE ENTERING THE TAIL CONE, FIRST SUPPORT THE AIRCRAFT TAIL WITH A SUITABLE TAILSTAND.

At the oxygen cylinder:

1) Remove screws attaching finished bulkhead to fuselage bulkhead.
2) Remove finished bulkhead.
3) With immediate area clear of flammables (grease, hydraulic fluid, fuel, etc.) and oxygen system off, connect a mask or tube to an outlet to exhaust any pressure in the system.

NOTE: Continuous pressure is applied to high pressure line until it is disconnected from cylinder. A check valve will close when high pressure line is disconnected from cylinder. The closing of this valve is frequently accompanied by a loud popping sound.

4) Unscrew the transducer from the branch tee fitting. Cap the opening in the branch tee.

(b) Installation

1) Attach transducer to the branch tee fitting.
2) Inspect any fittings that have been separated for leaks.
3) Reinstall and secure finished bulkhead in aft cabin.
4) Remove tailstand from airplane.

I. Refilling

CAUTION: BEFORE SERVICING THE OXYGEN SYSTEM, MAKE SURE THE AIRCRAFT IS SECURELY GROUNDED ELECTRICALLY.

CAUTION: DO NOT OPERATE ELECTRICAL EQUIPMENT WHILE SERVICING OXYGEN SYSTEM.

CAUTION: DO NOT ATTEMPT TO TIGHTEN ANY CONNECTIONS WHILE THE SYSTEM IS CHARGED.

Refilling of oxygen systems should be done by qualified personnel. For comparison of filling pressures to ambient temperatures refer to “Chart 3”. The following are parameters to be followed for filling.

(1) Only aviators breathing oxygen (MIL-PRF-27210) and appropriate filling equipment should be used to fill the system.

(2) If a cylinder has less than 5 psi pressure or has insufficient pressure to produce an audible hissing sound when the valve is cracked, it should be removed and/or purged, and if the condition has existed for a significant length of time, hydrostatically test cylinder.

(3) Make sure both the charge valve and recharge cart fittings are clean and free of contamination.

WARNING: BE CERTAIN THERE IS NO OIL OR OTHER PETROLEUM BASED MATERIAL ON THE FITTINGS OR IN THE IMMEDIATE VICINITY.

(4) Attach service cart hose to recharge port. Fill the system at a rate not exceeding 200 psig per minute proceeding as follows:

(a) To obtain the correct filling pressure for the oxygen system at various ambient temperatures, a table is included for your convenience. The pressures given are not exact, but sufficiently accurate for practical purposes of working pressures between 1800 and 2400 psig cylinders. The cylinder should be allowed to cool to a stabilized temperature after filling before checking against the values in “Chart 3”.

CHART 3
FILLING PRESSURES* FOR CERTAIN AMBIENT TEMPERATURES

Ambient Temperature °F/°C	Filling Pressure	Ambient Temperature °F/°C	Filling Pressure
0 / -17.78	1650 (PSI)	70 / 21	1975 (PSI)
10 / -12.22	1700	80 / 27	2000
20 / - 6.67	1725	90 / 32	2050
30 / - 1.11	1775	100 / 38	2100
40 / 4.44	1825	110 / 43	2150
50 / 10	1875	120 / 49	2200
60 / 15.56	1925	130 / 54	2250

* Filling pressures are for 1850 PSI at 70°F (21.11°C). Table assumes 25°F (11.8°C) rise due to heat of compressor with max. fill rate.

(b) When using a recharge unit consisting of one supply cylinder, slowly open the valve of the supply unit and allow the oxygen to transfer.

(c) When using a recharge unit consisting of two or more supply cylinders (cascade storage system), it is recommended that the following procedure be used:

1) Before opening any valves, check the pressure remaining in the airplane's oxygen cylinder. If it is still partly charged, note the pressure indicated on the cylinder gauge. Then open and close each valve on the cascade storage system and determine which cylinder has the lowest pressure. When found, if this cylinder has a pressure lower than the oxygen cylinder in the aircraft, do not attempt using it for filling; use the storage cylinder that has a pressure higher than the aircraft's cylinder but lower than the others.

2) Open the valve on only the one storage cylinder with the lowest pressure. When the pressure indicated on the aircraft's oxygen gauge and charging gauge has become equal, close the valve of the storage cylinder, then go to the storage cylinder with the next higher pressure and repeat the procedure.

3) If after using the last storage cylinder, the aircraft's oxygen system is still not fully charged, a full storage cylinder should be put in place of a cylinder with the lowest pressure and used in the same manner.

4) A good amount of oxygen will remain in the large cylinders used in the cascade system after filling only one of the cylinders. This remaining oxygen will be at a pressure something less than the 1850 psi. This is not sufficient pressure to completely refill another aircraft cylinder, although it will refill several small cylinders.

5) It is not economical, even on a three or four cylinder cascade system, to begin recharging with oxygen at less than 300 psi pressure in the 300 cubic foot bank of cylinders. So use 300 cubic foot cylinders down to approximately 300 psi; then return for refilling. In two cylinder systems use to approximately 100 psi; then return for filling.

(d) When the pressure gauge on the recharge unit or in the aircraft reaches 1800 to 1850 psi, close the pressure regulator valve on the recharge unit. Disconnect the filler hose from the filler valve; replace the protective cap on the filler valve and close the access cover. Check the cylinder pressure according to"Chart 3" after the cylinder temperature stabilizes.

(5) After detaching the service cart, cap hose and fittings to prevent contamination.

(6) Perform a leak check of the high pressure lines and clean off solution afterwards. If solution is not properly cleaned off, corrosion may result.

CHAPTER

37

VACUUM

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CHAPTER 37

LIST OF EFFECTIVE PAGES

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CHAPTER 37 - VACUUM

TABLE OF CONTENTS

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GENERAL

1. Description

The vacuum system (see “Figure 1” on page 37002) consists of two continuously operating, engine-driven dry-air pumps; one rotating clockwise and one rotating counterclockwise. Either pump can independently support the system. Also included are two regulators, an air inlet filter, and a manifold that connects the autopilot, pilot’s attitude indicator, cabin pressure controller, vacuum solenoid valve, and optional copilot instruments.

If aircraft is equipped with a pneumatic deicing system, refer also to 30-10-00.

A. PA-46-350P

In S/N’s 4636001 thru 4636020 only, a vacuum gauge incorporating two red flow buttons is mounted on the pilot’s instrument panel. When both pumps are operating satisfactorily, neither flow button is visible. The left flow button will protrude should the clockwise rotating pump fail. The right flow button will protrude should the counterclockwise rotating pump fail. The total vacuum pressure, which is displayed on the vacuum gauge, is normally regulated between 4.8–5.2 inches of mercury (inHg).

In S/N’s 4636021 thru 4636374 only, the vacuum gauge has been incorporated into the Transicoil Electronic Module Instrument System (see 77-40-00). A transducer mounted on the rear of the forward pressure bulkhead provides input to the Vacuum Pressure Indicator (VAC) which is co-located with the Cylinder Head Temperature (CHT) indicator The VAC indicator provides an analog display in the 3.0–6.0 inHg range. Normal operating pressures are between 4.5–5.2 inHg or 4.8–5.2 inHg, depending on the gauge installed in the particular aircraft. Vacuum pressure may also be displayed digitally on the Quad Digital Indicator/Enhanced Digital Indicator by pressing the function switch next to the analog display.

In S/N’s 4636375 and up, the vacuum display has been incorporated into the MFD.

(1) In Avidyne Entegra-equipped aircraft, a transducer mounted on the rear of the forward pressure bulkhead provides input to the Data Acquisition Unit (DAU) (see 77-40-00).

(2) In G1000/G1000 NXi-equipped aircraft, the vacuum transducer is mounted on the forward side of the forward pressure bulkhead. Data is sent from the vacuum transducer to the GEA, then to the #1 GIA and #2 GIA, then to the PFD1 and PFD2 (see 77-40-00). Nominal pressure is 5.0 inHg.

B. PA-46R-350T

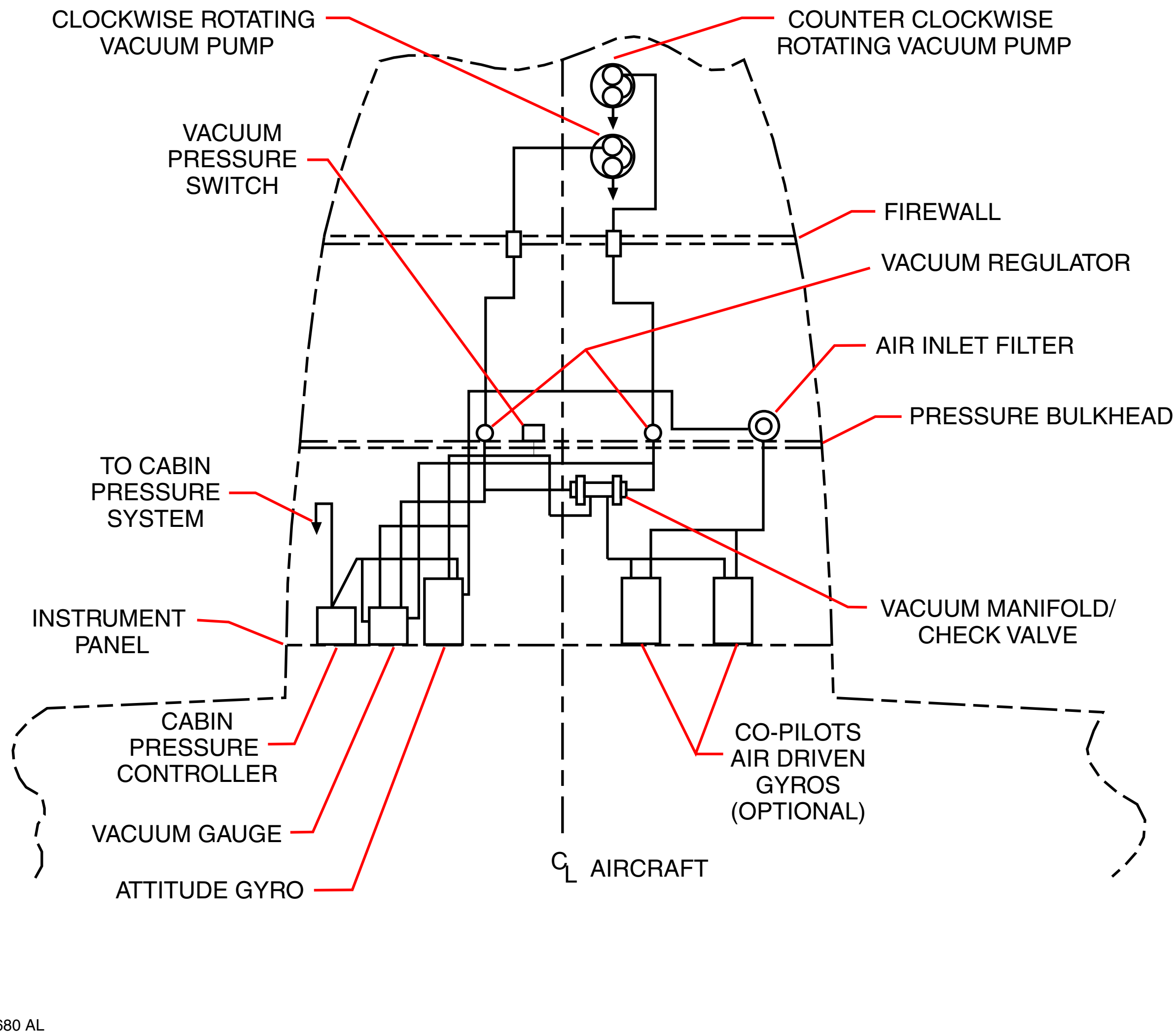
In S/N’s 4692001 and up, the vacuum display (if installed) has been incorporated into the MFD.

(1) In Avidyne Entegra-equipped aircraft, a transducer mounted on the rear of the forward pressure bulkhead provides input to the Data Acquisition Unit (DAU) (see 77-40-00).

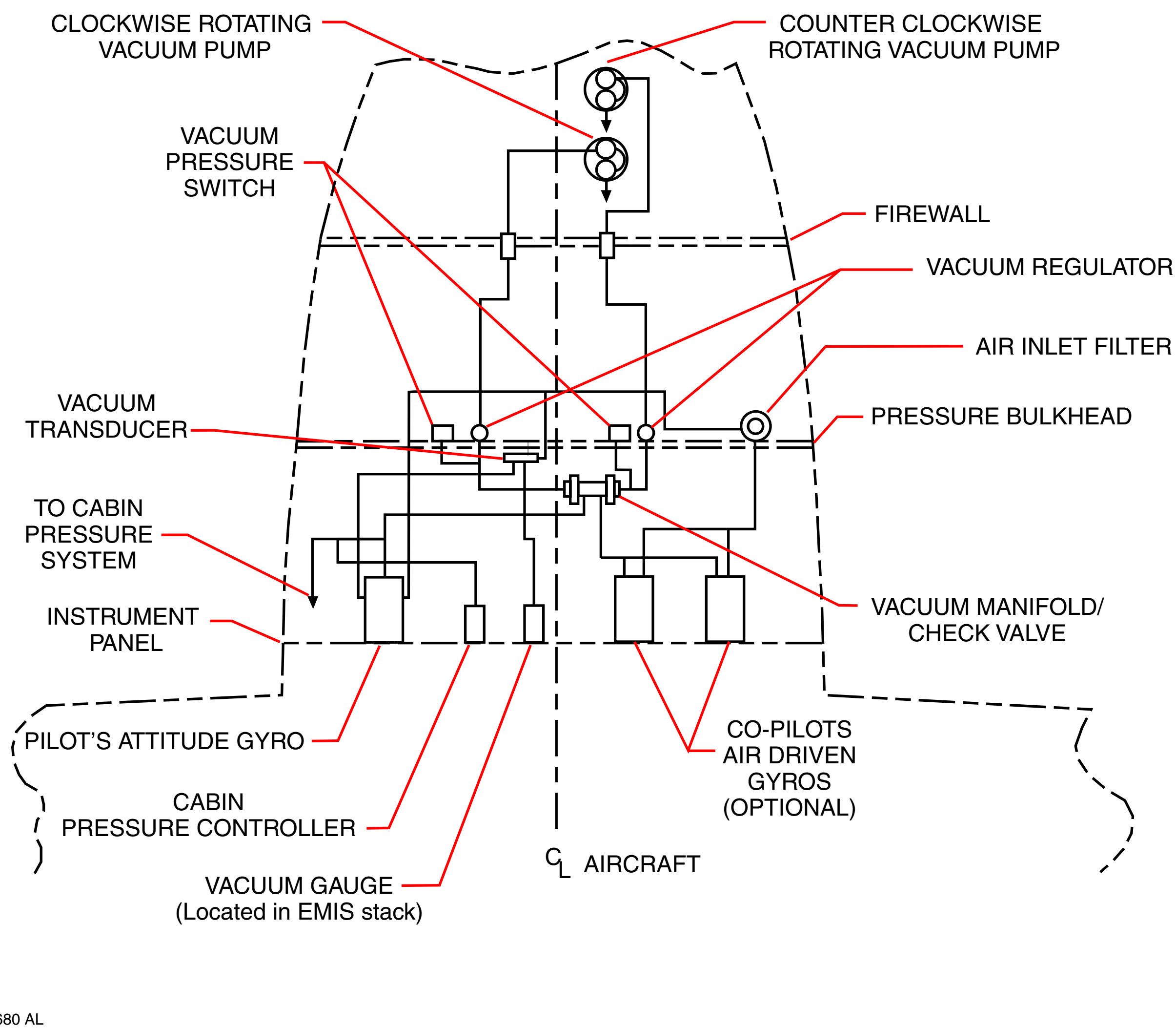
(2) In G1000-equipped aircraft, the vacuum transducer is mounted on the forward side of the forward pressure bulkhead. Data is sent from the vacuum transducer to the GEA, then to the #1 GIA and #2 GIA, then to the PFD1 and PFD2 (see 77-40-00). Nominal pressure is 5.0 inHg.

2. Troubleshooting

See “Chart 1” on page 37006.

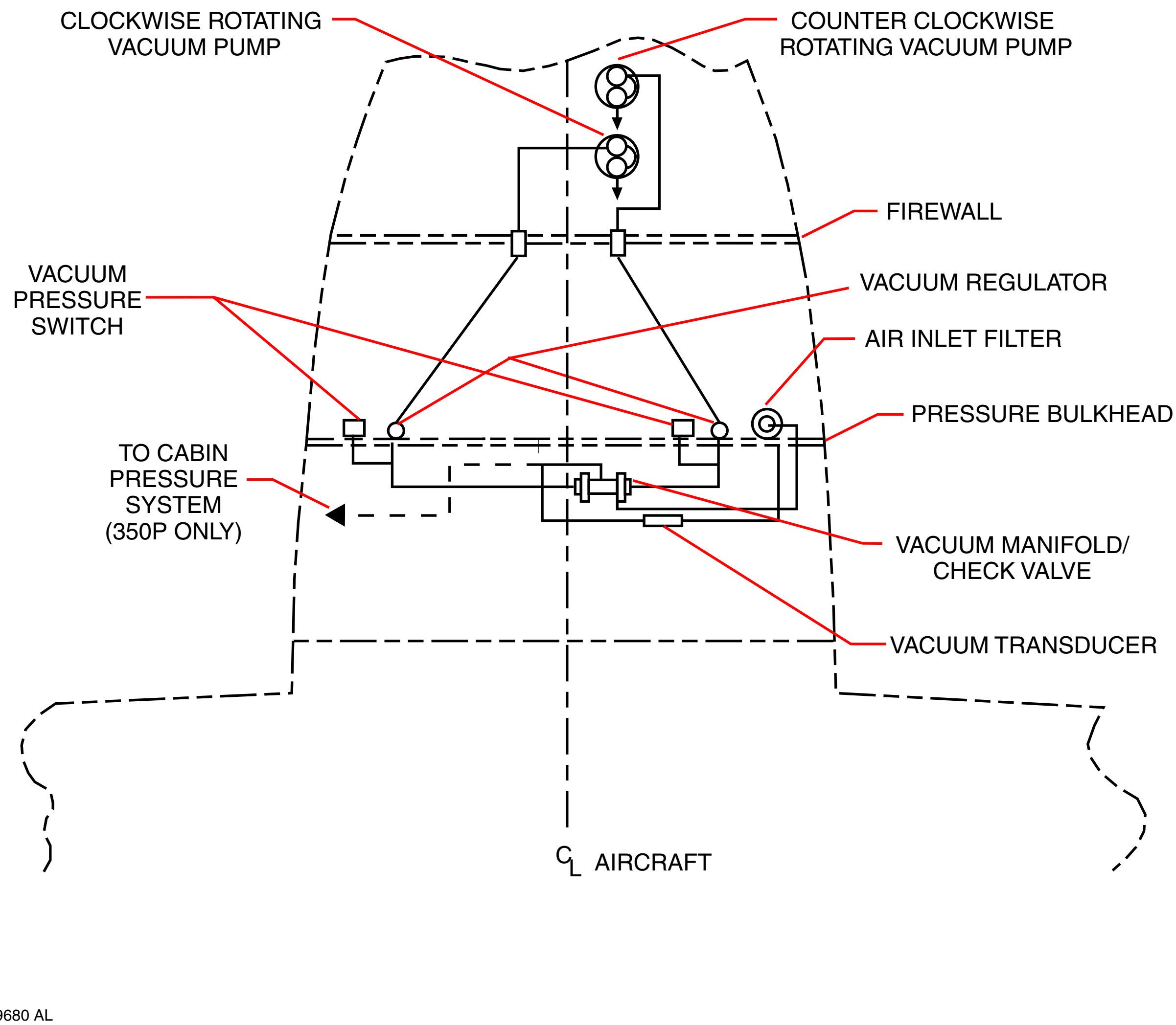


Vacuum System
Figure 1 (Sheet 1 of 4)



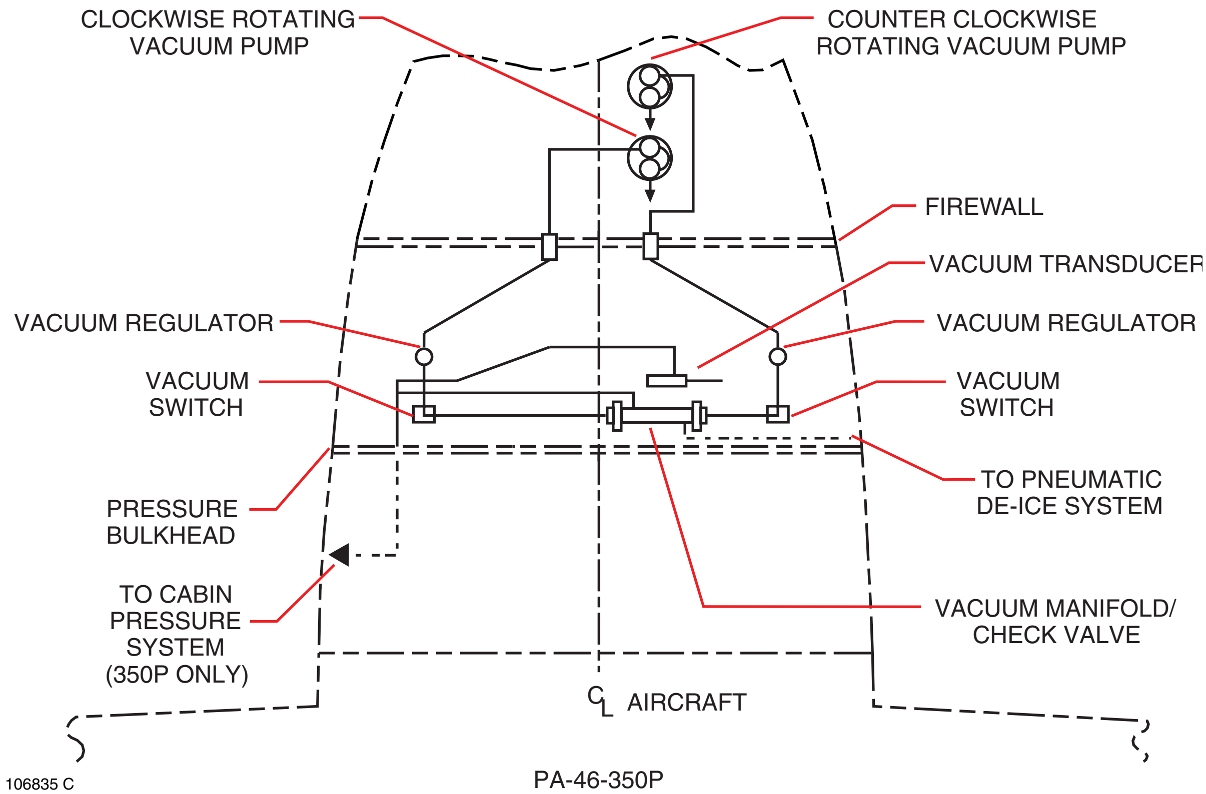
Vacuum System
Figure 1 (Sheet 2 of 4)

Effectivity
4636021 thru 4636374

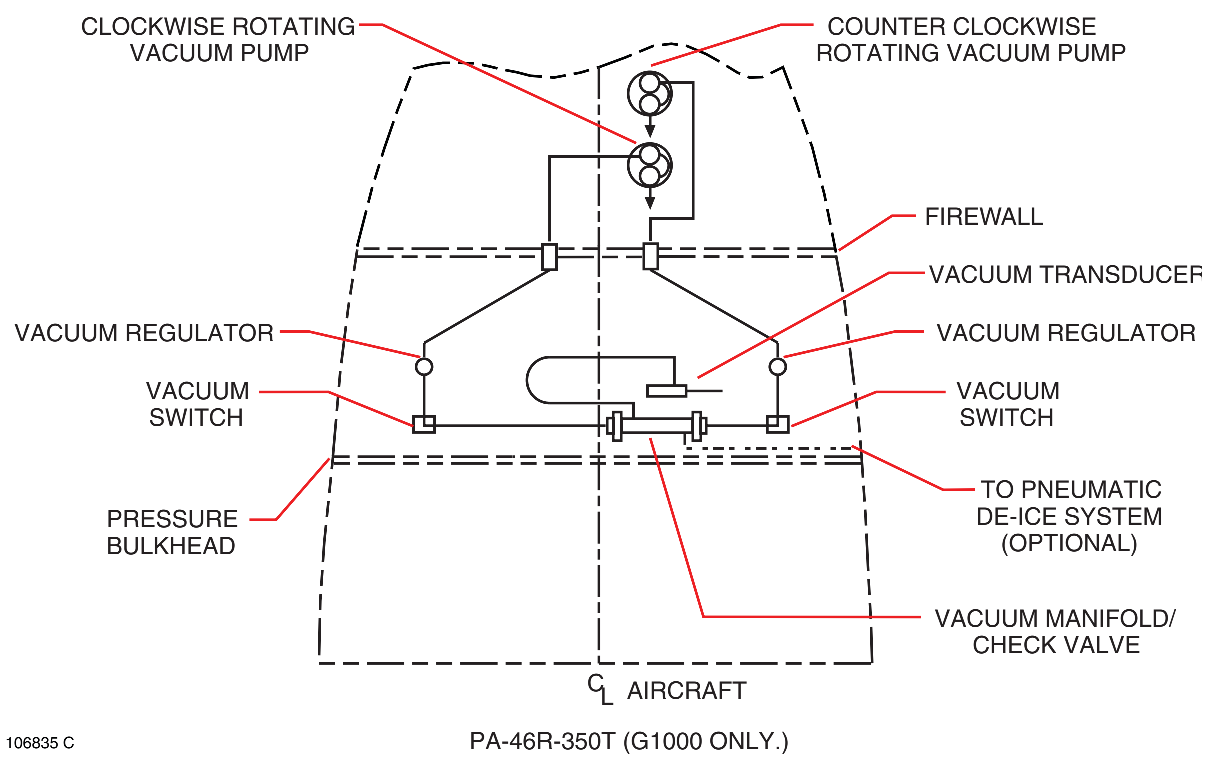


Effectivity
4636375 and up
4692001 and up

Vacuum System
Figure 1 (Sheet 3 of 4)



PA-46-350P



PA-46R-350T (G1000 ONLY.)

Effectivity with G1000/G1000 NXi installation

Vacuum System
Figure 1 (Sheet 4 of 4)

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING VACUUM SYSTEM
(for S/N's 4636021 thru 4636374, see also Chart 1 in 77-40-00)

Trouble	Cause	Remedy
Attitude gyro(s) not receiving sufficient vacuum for proper operation. Low reading on vacuum gauge.	Filter clogged or dirty.	Clean or replace filter.
	Line from gyro to filter restricted.	Check line.
No vacuum gauge indication, but gyro instrument(s) operating satisfactorily.	Faulty gauge.	Replace gauge.
Low vacuum system pressure indicated by low reading on vacuum gauge.	Filter dirty.	Clean or replace filter.
	Vacuum regulator valve incorrectly adjusted.	Adjust regulator valve in accordance with Adjustments in this section.
	Line from gyros to filter restricted.	Repair or replace line.
	Line between pumps and manifold, or between gyros and manifold leaking.	Check all lines and fittings.
In S/N's 4636001 thru 4636020 only:		
Vacuum gauge reading 4.8 to 5.2 in. Hg, but left red button on gauge protruding	Clockwise rotating vacuum pump inoperative.	Replace lower vacuum pump.
or		
Vacuum gauge reading 4.8 to 5.2 in. Hg, but right red button on gauge protruding.	Counterclockwise rotating vacuum pump inoperative.	Replace upper vacuum pump.
Zero vacuum gauge reading.	Both pumps faulty.	Replace pumps.
Both red buttons protruding.	Faulty vacuum pressure switch.	Replace switch.
Normal pressure indication, but sluggish operation of instruments.	Faulty instrument.	Replace instrument.
High system pressure.	Vacuum regulator incorrectly adjusted.	Adjust regulator.
	Vacuum regulator sticking or dirty screen.	Clean and check operation of regulator.
NOTE: The vacuum is displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.		

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING VACUUM SYSTEM
(for S/N's 4636021 thru 4636374, see also Chart 1 in 77-40-00)

Trouble	Cause	Remedy
Regulator cannot be adjusted to produce correct pressure.	Lines leaking.	Check lines and fittings.
	Vacuum pump(s) malfunctioning.	Replace pump(s).
Vacuum correct on ground, but will not maintain pressure at altitude.	Vacuum pump(s) malfunctioning.	Replace pump(s).
	Regulator sticky.	Clean regulator.
Vacuum correct, but pilot reports pressure erratic or shows complete loss in flight.	Regulator sticky.	Clean regulator.
	Oil in pump due to leaky engine seal or cleaning fluid blown into pump while cleaning engine.	Replace pump.
Pressure can only be maintained at full throttle on ground.	Leak in system.	Repair or replace lines.
	Worn pump.	Replace pump.
	Stuck regulator.	Clean or replace regulator.
Gauge indication follows engine rpm.	Foreign matter on regulator(s) seat.	Raise regulator(s) diaphragm with thin, blunt tool (tongue depresser). Remove the contaminant. Reset regulator. If no foreign matter is present on the regulator seat, test regulator function.

NOTE: The vacuum is displayed on the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed.

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DISTRIBUTION

The following information is intended to acquaint field service personnel with a means to diagnose vacuum system service symptoms on those components which are serviced by removal and replacement. These items include hoses, clamps, gyro filters. vacuum-regulating valves, and vacuum gauges.

1. Hoses and Clamps

A. These items should be examined periodically and inspected carefully whenever maintenance activities cause hose disconnections to be made at the pumps, manifold, regulators, tube assemblies, gyros and/or vacuum gauge.

B. Ends of hoses should be examined for rubber separation and slivers of rubber on inside diameter of hoses. These slivers can and do become detached. If this happens, the vacuum pump(s) will suck in the loose particles and eventually ingest them. This can cause pump failure.

C. Replace old, hard, cracked or brittle hose. Sections of the inner layers may separate, causing pump failure.

D. Ensure hoses are clear and clean by blowing them out with shop air. Remove from aircraft as required.

CAUTION: DO NOT WIGGLE HOSE FROM SIDE TO SIDE DURING INSTALLATION. WIGGLING COULD CAUSE PARTICLES TO BE CUT FROM INNER WALL OF HOSE WHICH WOULD DAMAGE THE PUMP.

E. Where hose clearance is tight, making it difficult to reinstall it onto a fitting or barb, spray the fitting or barb with silicone. Let dry, then install hose by pushing it straight on.

CAUTION: WHEN REPLACING ANY OF THE THREADED FITTINGS, DO NOT USE PIPE DOPE, THREADLUBE, OR TAPE. PIPE DOPE / TAPE PARTICLES INGESTED BY THE VACUUM PUMP COULD CAUSE THE PUMP TO FAIL. USE ONLY SILICONE SPRAY, LETTING IT DRY BEFORE ASSEMBLY.

F. Hose clamps and fittings should be replaced when broken, damaged or corroded.

2. Gyro Filter

NOTE: Applies to aircraft with conventional instruments or Avidyne installation. Not applicable to G1000/G1000 NXi installation.

A. Gyro filters must be serviced on a scheduled basis, not to exceed 100 hours, or sooner as condition indicates.

B. The system installation employs a large central filter and differential vacuum gauge that continuously monitors the filter condition while indicating vacuum readings.

NOTE: A decline in panel gauge reading indicates the filter is becoming clogged. Filters should be replaced when gauge reading declines; DO NOT adjust regulator(s).

3. Vacuum Pump(s)

Two (2) engine-driven dry-air pumps are mounted on the accessory section at the rear of the engine. Each vacuum pump is a rotary vane, positive displacement type. These units consist of an aluminum housing, a carbon rotor and carbon vanes. They are driven by means of a coupling mated to an engine-driven gear assembly.

WARNING: GROUND THE MAGNETO PRIMARY CIRCUIT (P LEAD), BEFORE PERFORMING ANY MAINTENANCE OPERATION ON THE ENGINE.

A. Inspection

Aero Accessories Tempest dry air pumps are factory installed in S/N's 4636339 and up and 4692001 & up, and as service replacements. If so equipped; (see "Figure 1")

- for airplanes with frequent de-ice cycles, beginning at 200 hours time-in-service, and each 100 hours thereafter;
- for airplanes with normal de-ice cycles, beginning at 300 hours time-in-service, and each 100 hours thereafter;

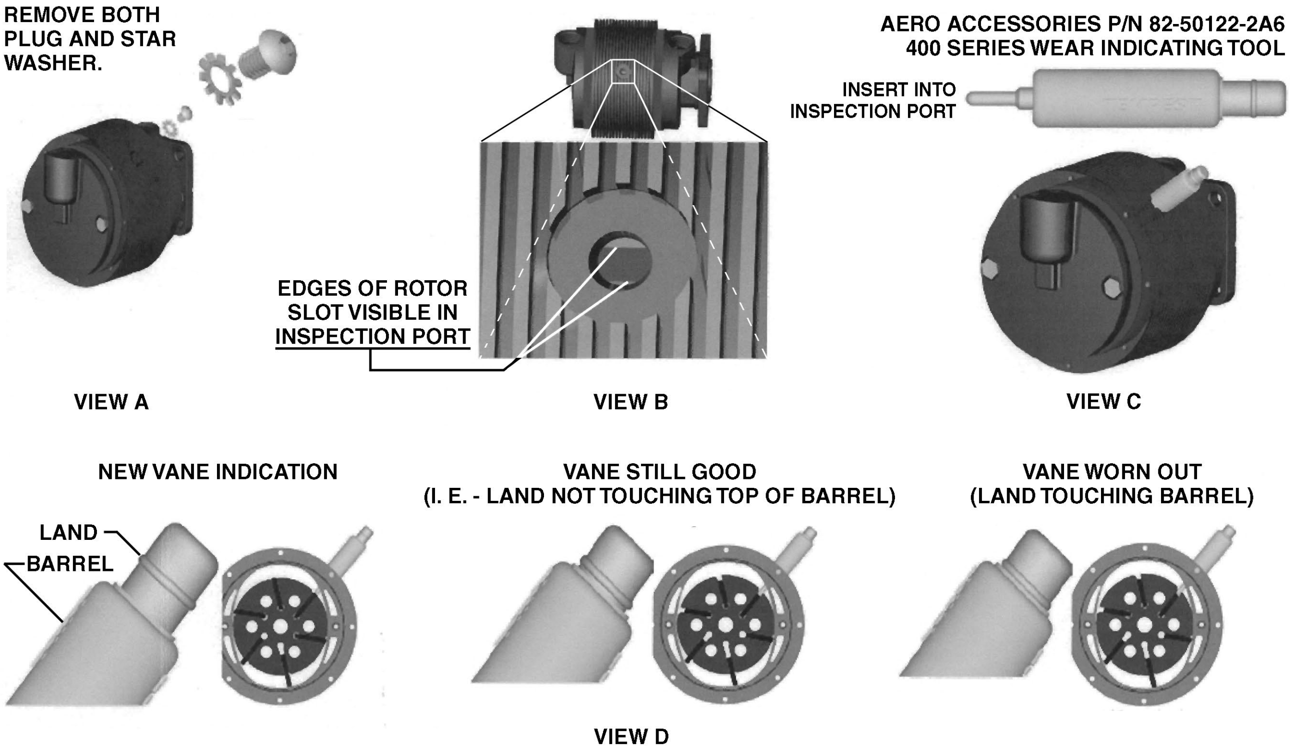
inspect the vacuum pumps as follows:

(1) Remove engine cowling. (See 71-10-00.)

(2) Ground magnetos and turn the fuel selector to OFF.

CAUTION: DIRT, GREASE, OR DEBRIS IN THE AREA OF THE INSPECTION PORT PLUG COULD CONTAMINATE THE PUMP IF THE PLUG IS REMOVED PRIOR TO CLEANING.

(3) Ensure the area around the inspection port plug is clean.



Vacuum Pump Vane Wear Measurement
Figure 1

37102

CAUTION: IF THE STAR WASHER IS NOT REMOVED WITH THE PLUG, THE SUBSEQUENT WEAR INDICATION READING WILL BE ERRONEOUS.

(4) Remove the inspection port plug and star washer (View A).

(5) While looking into the inspection port, have an assistant slowly turn the propeller by hand in the normal direction of rotation until a vane slot is centered in the port (View B).

NOTE: If you go too far, just keep turning the propeller until the next vane slot appears. DO NOT turn the propeller backwards.

CAUTION: DO NOT ROTATE PROPELLER OR PUMP SHAFT WITH THE INDICATING TOOL INSERTED INTO THE INSPECTION PORT. IF THE PUMP IS INADVERTENTLY ROTATED WHILE THE INDICATING TOOL IS INSERTED INTO THE INSPECTION PORT, PUMP REPLACEMENT IS REQUIRED - EVEN IF YOU CAN DETECT NO DAMAGE.

(6) Insert the vane wear indicating tool into the inspection port as illustrated in View C.

(a) Hold the barrel securely and squarely against the pump body.

(b) With your finger tip, gently push the plunger into the pump. When the plunger touches the vane you may feel the vane move slightly if it is not at the bottom of its slot.

NOTE: If the plunger does not slip easily into the slot, DO NOT force it. Remove the indicating tool and check the alignment of the vane slot to the inspection port.

(7) Compare the relative positions of the plunger land and barrel top to the illustrations in View D to determine vane wear.

(a) If vane is “worn out,” replace pump.

(b) If vane is “still good” or “new,” proceed to next step.

(8) If vanes are within service limits and pump is otherwise serviceable, clean the threads on the inspection port plug, install a new star washer and torque the port plug to 45 to 50 in.-lbs.

B. Removal

(1) Remove engine cowling. (Refer to Chapter 71.)

(2) Loosen hose clamp and remove hose from pump fittings.

(3) Remove four retaining nuts, lock washers and plain washers used to secure pump to engine; then remove pump.

C. Installation

WARNING: WHEN INSTALLING THE LOWER VACUUM PUMP, ENSURE THE FUEL INJECTOR LINKAGE ROD AND THE LINKAGE ATOP THE VARIABLE PRESSURE CONTROLLER BOTH MOVE FREELY AND CLEARLY THROUGHOUT THEIR FULL RANGE OF MOTION. SPECIFICALLY LOOK FOR INTERFERENCE BETWEEN THE LOWER VACUUM PUMP COOLING SHROUD / BLAST HOSE AND THE FUEL INJECTOR LINKAGE ROD AND/OR THE LINKAGE ATOP THE VARIABLE PRESSURE CONTROLLER.

NOTE: Change the vacuum system filter when installing a new pump.

(1) If required, install fittings on pump per Replacing Pump Fittings, below.

CAUTION: ONLY PUMP MOUNTING GASKET AUTHORIZED AND APPROVED FOR USE ON AIRBORNE VACUUM PUMP IS AIRBORNE GASKET B3-1-2, PIPER PART NUMBER 751-859. USE OF ANY OTHER GASKET MAY RESULT IN OIL SEEPAGE OR LEAKAGE AT MOUNTING SURFACE.

(2) Place pump gasket in its proper place and align spline on pump drive with spline on engine drive assembly.

(3) Secure pump to engine with four plain washers, lock washers and retaining nuts. Torque nuts 50 to 70 inch-pounds.

(4) Connect hoses to pump and secure with hose clamps.

(5) Reinstall engine cowling.

D. Replacing Pump Fittings

CAUTION: WHEN REPLACING ANY OF THE THREADED FITTINGS, DO NOT USE PIPE DOPE, THREADLUBE, OR TAPE. PIPE DOPE / TAPE PARTICLES INGESTED BY THE VACUUM PUMP COULD CAUSE THE PUMP TO FAIL. USE ONLY SILICONE SPRAY, LETTING IT DRY BEFORE ASSEMBLY.

(1) Before installing any fittings on pump, check for any external damage. A pump that has been damaged or dropped should not be installed.

CAUTION: DO NOT APPLY VISE PRESSURE TO OUTSIDE DIAMETER OR OVERALL LENGTH OF PUMP.

(2) When a vise is used to hold pump while installing fittings, suitable caution must be exercised to avoid pump damage. Square mounting flange must be held between soft wood blocks and only at right angles to vise jaws. Use only enough vise pressure to hold pump firmly.

(3) The ports of AIRBORNE pumps have been treated with a dry film lubricant and AIRBORNE fittings are cadmium plated thus eliminating any need for thread lubricants. If thread lubricant is required, use only a silicone spray. Apply sparingly to external threads of fittings only and let dry before assembly.

4. Vacuum Regulator(s)

Two (2) vacuum regulators are incorporated in the system to control vacuum pressure to gyro instruments. The regulators are located above the overhead trim panel in the forward baggage compartment.

A. Removal

(1) Remove inlet hose from aft side of valve.

(2) Remove outlet hose from forward side of valve.

(3) Remove nut that secures valve to firewall.

(4) Pull valve rearward out of firewall.

B. Installation

(1) Insert valve into hole in rear of firewall.

(2) Secure valve to firewall with nut.

(3) Attach inlet hose to forward fitting of valve.

(4) Attach outlet hose to aft fitting of valve.

C. Service Tips

(1) The vacuum-regulating valve seldom needs replacement. Symptoms that suggest replacement are:

(a) Chatter as indicated by rapid fluctuation of the vacuum gauge needle or an audible sound.

(b) Non-repeatability of the vacuum gauge reading when the panel gauge is not suspect or has been checked against a known test gauge (cruise rpm only).

(2) All modes of regulator malfunction tend to increase the vacuum power applied to the gyros. Thus, although excess vacuum is applied, a loss of vacuum does not occur.

(3) The gyros themselves act as a limiting device to keep the vacuum power applied from exceeding safe levels.

NOTE: If the panel gauge has been checked and found “OK” and the vacuum gauge reading does not repeat within the normal operating range as marked on the gauge, then the regulating valve should be changed. Observe the usual precautions for maintaining system cleanliness to avoid premature pump service.

D. Adjustment

NOTE: Do not reset the regulator until the filter and lines have been checked

(1) Disconnect inlet line going to one of the vacuum regulator valves.

(2) Wrap or cap the end of line with a clean, porous cloth to prevent foreign matter and debris from entering the system.

NOTE: Do not attempt to adjust these valves with the engine in operation without a qualified pilot or other responsible person at the controls of the airplane.

(3) Start the engine, allow it to warm up and run the engine at magneto check rpm.

(4) While running the engine at magneto check rpm, the suction gauge (or vacuum pressure indicator) should indicate within the normal operating range as marked on the gauge. If the reading is not in this range, shut down the engine and adjust the valve. (Turn the valve adjustment screw clockwise to increase pressure and counterclockwise to decrease pressure.)

(5) Start the engine and repeat the check.

(6) After the system has been properly adjusted, bend the locking tabs on the valve down over the adjustment screw.

(7) Uncap the line that goes to the vacuum regulator valve and connect it to the valve.

(8) Disconnect line going to the other vacuum regulator valve.

(9) Wrap or cap the end of line with a clean, porous cloth to prevent foreign matter and debris from entering the system.

(10) Repeat Steps (3) through (7).

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INDICATING

1. Vacuum (Suction) Gauge

S/N's 4636001 thru 4636020 only.

NOTE: S/N's 4636021 thru 4636374 incorporate Integrated Engine Instrumentation Systems. Refer to Section 77-40-00 for more information.

NOTE: S/N's 4636375 and up incorporate the vacuum display in the MFD when either Avidyne Entegra or Garmin G1000/G1000 NXi is installed. Refer to Section 77-40-00 for more information.

NOTE: S/N's 4692001 and up incorporate the vacuum display (if installed) in the MFD when either Avidyne Entegra or Garmin G1000 is installed. Refer to Section 77-40-00 for more information.

The vacuum (suction) gauge is mounted on the left-hand side of the instrument panel. This gauge is calibrated in inches of mercury, and indicates the amount of vacuum created by the engine-driven vacuum pumps. The total vacuum pressure, which is displayed on the vacuum gauge, is normally regulated between 4.8–5.2 inches of mercury (inHg).

The vacuum gauge also incorporates two red flow buttons. When both pumps are operating satisfactorily, neither flow button is visible. The left flow button will protrude should the clockwise (lower) rotating pump fail. The right flow button will protrude should the counterclockwise (upper) rotating pump fail.

If the system filter becomes clogged or, as the lines become obstructed, the gauge will show a decrease in pressure. Do not reset the regulator until the filter and lines have been checked. Should the vacuum pressure from any one pump fall below the minimum pressure required, its red button will protrude from the instrument face.

A. Service Tips

(1) The suction gauge seldom requires service and usually is replaced when malfunctions occur.

NOTE: Suction gauge failure in a properly operating vacuum system does not impair safety of flight.

NOTE: Two different models of vacuum gauge may be installed in PA-46-350P airplanes. One is marked with a normal operating range of 4.8–5.2 inHg and the other is marked with a normal operating range of 4.5–5.2 inHg. The vacuum system in a particular airplane should be adjusted to operate in the pressure range indicated on the gauge installed in that aircraft.

(2) If the suction gauge malfunctions in a manner to cause an incorrect reading in normal cruise conditions, the gauge must be checked by comparing the reading with a gauge of known accuracy. If the gauge is indicating correct values and the system vacuum level is not in accordance with the specified vacuum, then, and only then, should the regulator be reset.

(3) Visual examination of the gauge performance should cover the following steps:

(a) With engine stopped and no vacuum supplied to the gauge, its pointer should rest against the internal stop in the 9 o'clock position. Any displacement from this position suggests need for replacement.

(b) A slight overshoot during engine start-up, not to exceed half an inch of Hg. (1/2 inHg), is normal and is not cause to replace gauge.

(c) The gauge should read in the normal operating range as marked on the vacuum gauge with engine operating at normal cruise rpm.

(d) At 1,200 rpm, the vacuum gauge reading should be more than four inches of mercury.

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CHAPTER

39

ELECTRICAL / ELECTRONIC PANELS

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CHAPTER 39

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
39 - List of Effective Pages	1	Mar 25/21
	2	Sep 15/09
39 - Table of Contents	1	Mar 25/21
	2	Sep 15/09
39-10-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Nov 30/17
	5	Mar 25/21
	6	Nov 30/17
	7	Nov 30/17
	8	Nov 30/17
	9	Nov 30/17
	10	Nov 30/17
	11	Nov 30/17
	12	Sep 15/09
39-20-00	1	Dec 15/14
	2	Dec 15/14
	3	May 31/11
	4	Dec 15/14
	5	Dec 15/14
	6	Dec 15/14
	7	Dec 15/14
	8	Dec 15/14
	9	Dec 15/14
	10	Dec 15/14
	11	Mar 25/21
	12	Mar 25/21
	13	Mar 25/21
	14	Mar 25/21
	15	Mar 25/21
	16	Mar 25/21
	17	Mar 25/21
	18	Mar 25/21
	19	Mar 25/21
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	27	Mar 25/21
	28	Mar 25/21
	29	Mar 25/21
	30	Mar 25/21
	31	Mar 25/21
	32	Mar 25/21
	33	Mar 25/21
	34	Mar 25/21
	35	Mar 25/21
	36	Mar 25/21

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CHAPTER 39 - ELECTRICAL / ELECTRONIC PANELS

TABLE OF CONTENTS

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INSTRUMENT AND CONTROL PANELS

1. General

A. Face-Mounted Instruments

Most instruments are face-mounted and secured to the instrument panel by screws from the front ofthe panel. Most instruments are removed out the back of the panel, but a few are removed through the front of the panel. Take special care when any operation pertaining to the instruments is performed.

(1) Removal

(a) Disconnect the plumbing and / or electrical connectors from the back of the instrument. Where two or more lines connect to an instrument, identify and tag each line to facilitate installation. Attach a dust cap to each fitting.

NOTE: For those instruments which remove through the front of the panel, disconnecting and tagging plumbing and / or electrical connectors can be done after the instrument retaining screws are removed and the instrument is slid gently forward to expose the connections at the rear.

(b) Remove the screws that secure the instrument in the panel cutout.

(c) Remove the instrument from the panel.

(2) Installation

(a) Place the instrument in its proper panel cutout and secure with screws.

NOTE: For those instruments which install through the front of the panel, connecting plumbing and / or electrical connectors can be done from the front of the panel before the instrument retaining screws are installed. After the connections are secure, slide the instrument into place and install the retaining screws.

(b) Connect the plumbing and/or electrical connectors to back of instrument.

(c) Check instrument operation.

B. Rack-Mounted Avionics

(PIR-PPS60237, Rev. New.)

Most avionics are rack-mounted, front-removable units, generally secured to the instrument panel tray/rack by a single jackscrew located in the center of their faceplate.

(1) Removal

(a) Insert an appropriate size (generally 3/32 inch) allen wrench into the jackscrew access hole in the faceplate.

(b) Unscrew the jackscrew in a counterclockwise direction.

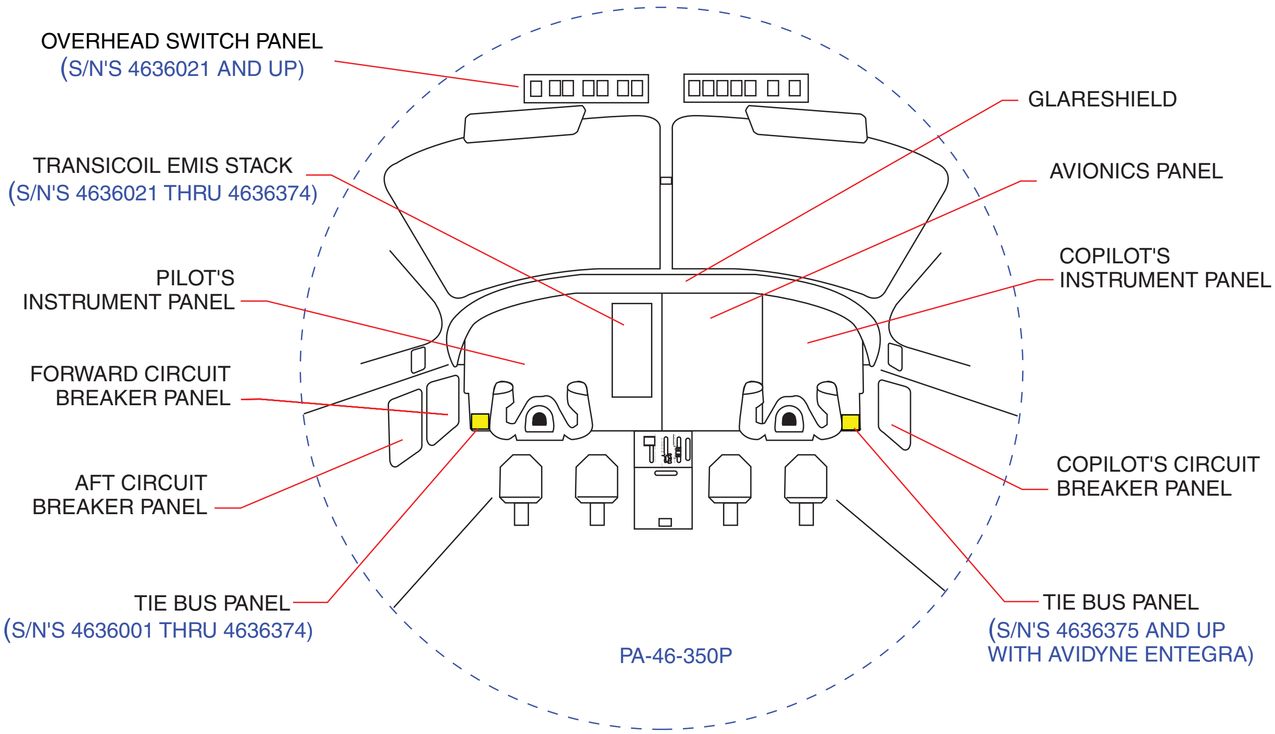
(c) Slide the avionics unit aft and out of the instrument panel tray/rack.

(2) Installation

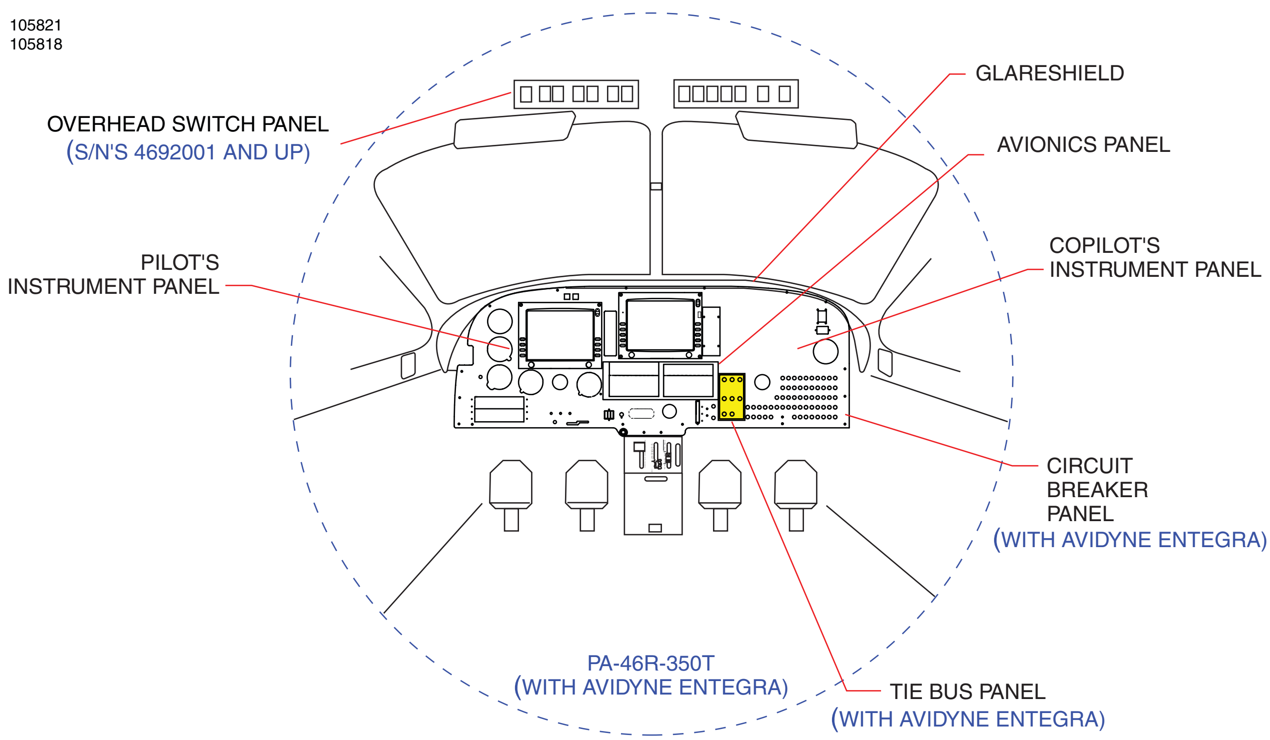
NOTE: Inspect the front of the panel-mounted avionics tray/rack to verify it is not significantly inset from the panel. If so, correct the tray/rack installation before proceeding.

NOTE: The high insertion forces required to seat a unit with “high density” connectors tend to limit the effectiveness of the first seating attempt. Accordingly, the following procedure requires sequential applications of force, and subsequent tightening of the jackscrew, to ensure all connectors seat properly.

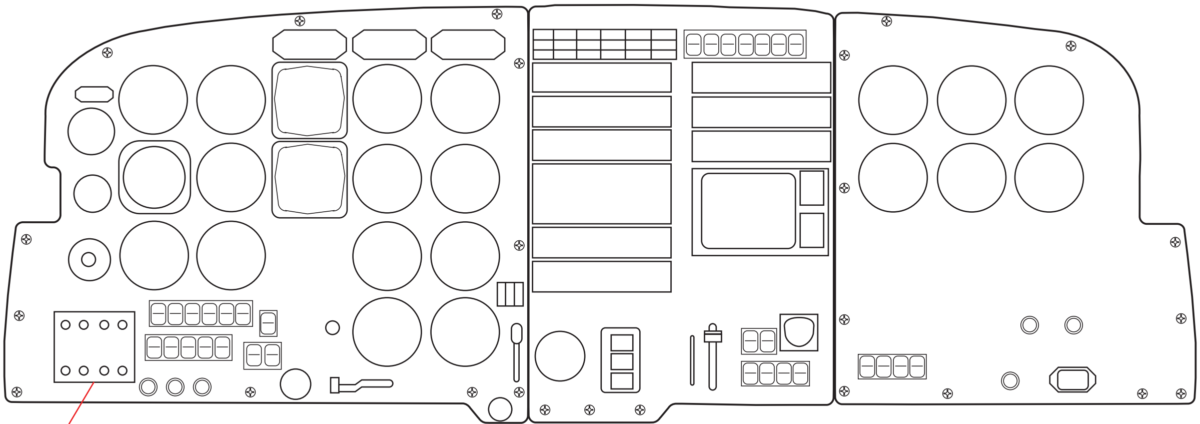
(a) Slide the avionics unit into the instrument panel rack and forward, applying a moderate insertion force.



105821
105818

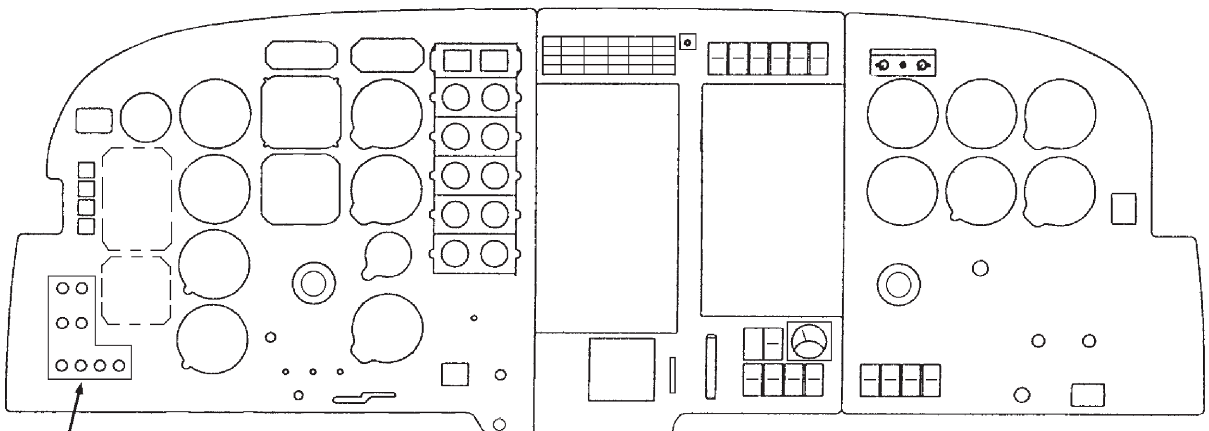


Flight Deck View (Typical)
Figure 1



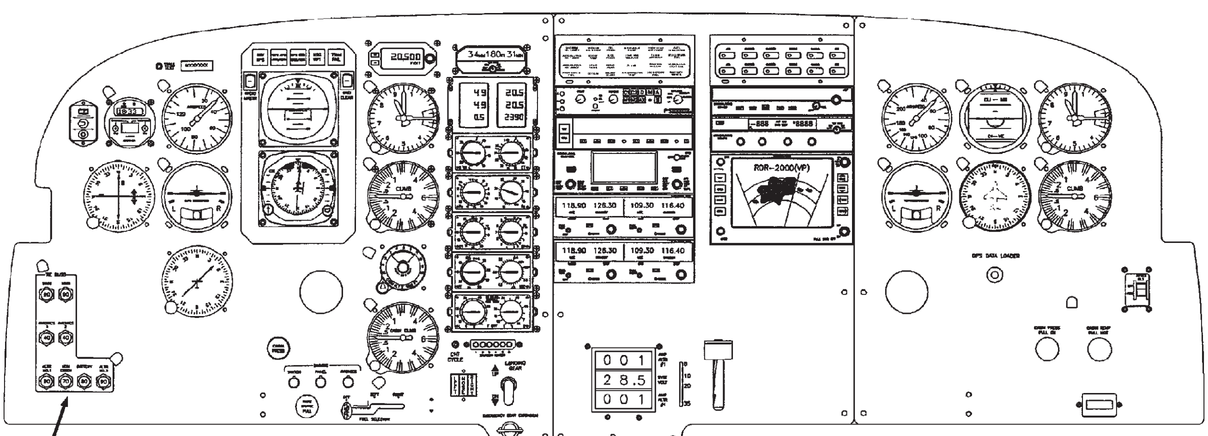
TIE BUS
CIRCUIT BREAKER PANEL

(S/N's 4636001 THRU 4636020 ONLY)



TIE BUS
CIRCUIT BREAKER PANEL

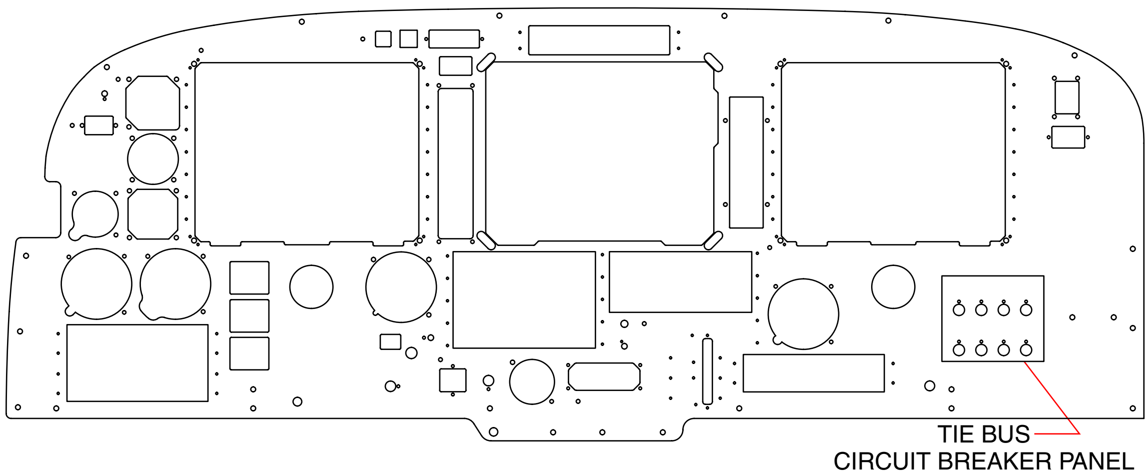
(S/N'S 4636021 THRU 4636131 ONLY)



TIE BUS
CIRCUIT BREAKER PANEL

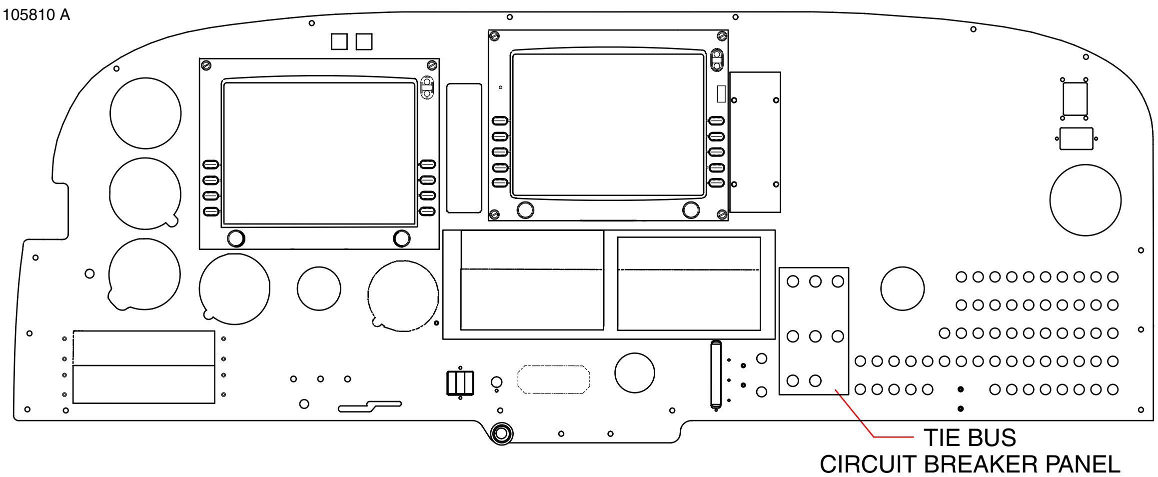
(S/N'S 4636132 THRU 4636374 - STANDARD)

Instrument Panel Installations (Typical)
Figure 2 (Sheet 1 of 3)



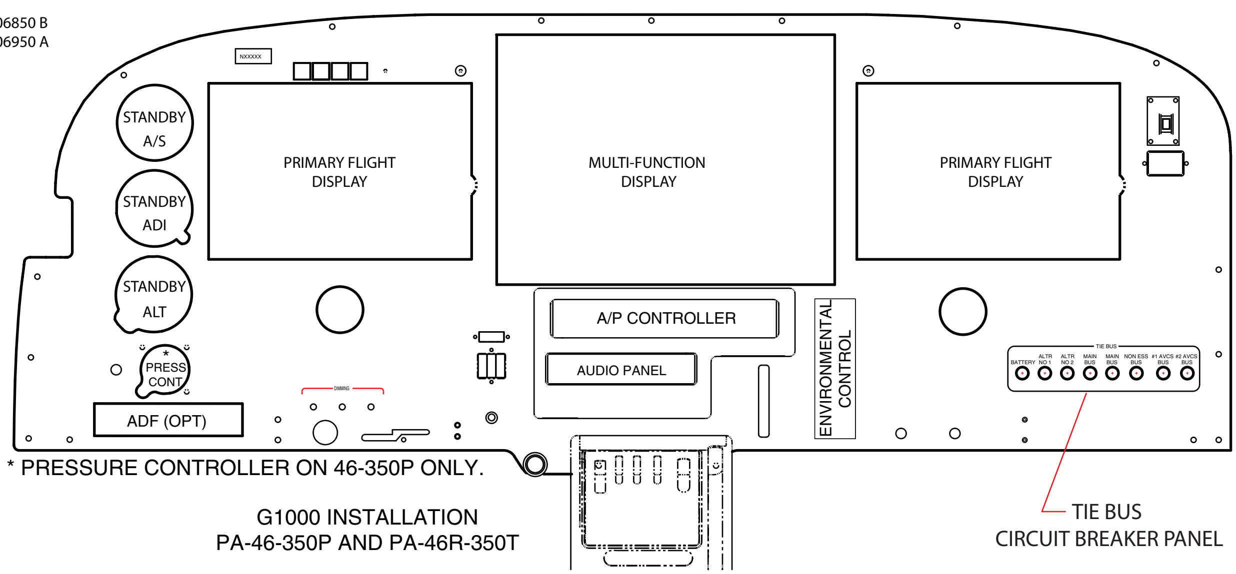
(S/N's 4636375 AND UP WITH AVIDYNE ENTEGRA)

105810 A



(S/N's 4692001 AND UP WITH AVIDYNE ENTEGRA)

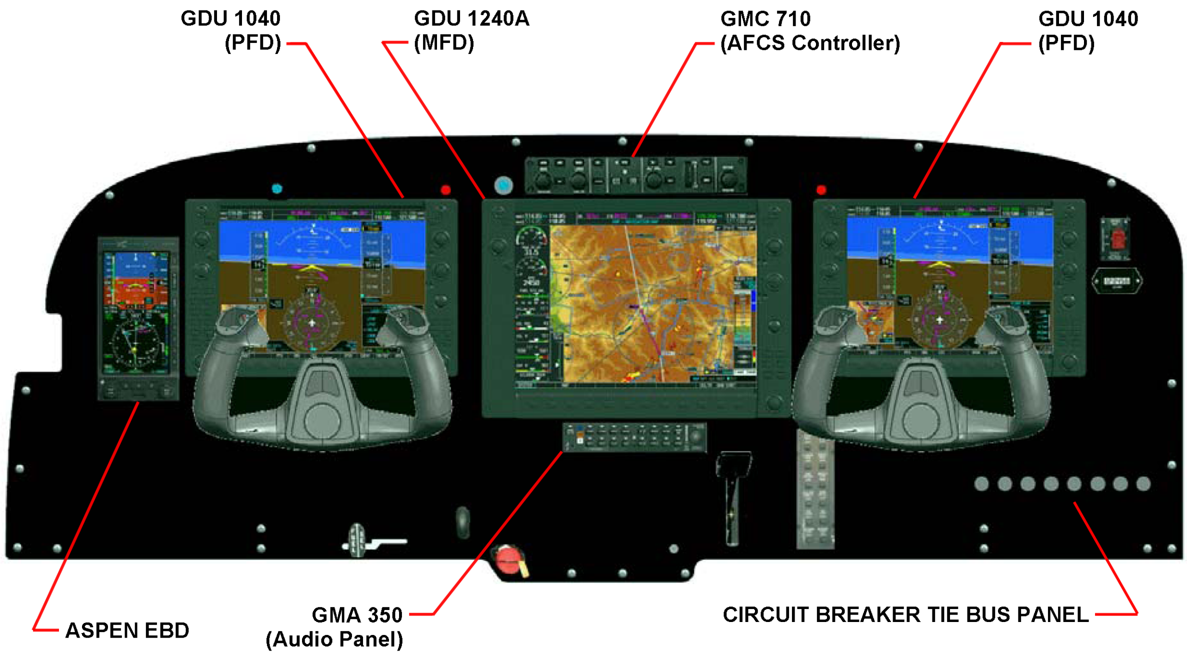
106850 B
106950 A



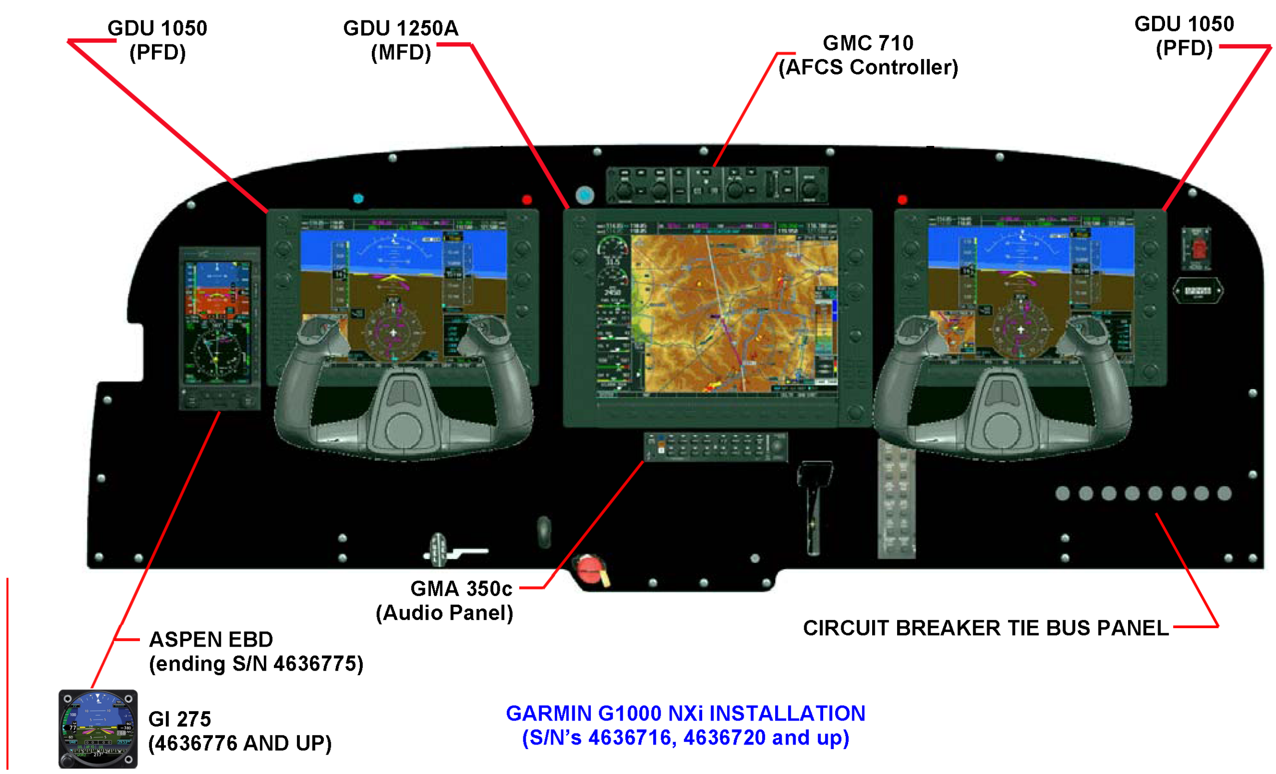
* PRESSURE CONTROLLER ON 46-350P ONLY.

G1000 INSTALLATION
PA-46-350P AND PA-46R-350T

Instrument Panel Installations (Typical)
Figure 2 (Sheet 2 of 3)



GARMIN G1000 INSTALLATION
(S/N's 4636633, 4636652–4636715, 4636717–4636719)



GARMIN G1000 NXi INSTALLATION
(S/N's 4636716, 4636720 and up)

Instrument Panel Installations (Typical)
Figure 2 (Sheet 3 of 3)

(b) Insert an appropriate size (generally 3/32 inch) allen wrench into the jackscrew access hole in the faceplate and tighten to remove any slack, but do not try to “pull” unit into place with the jackscrew.

(c) Apply additional insertion force to front of unit.

(d) Tighten jackscrew again.

(e) Apply additional insertion force to front of unit.

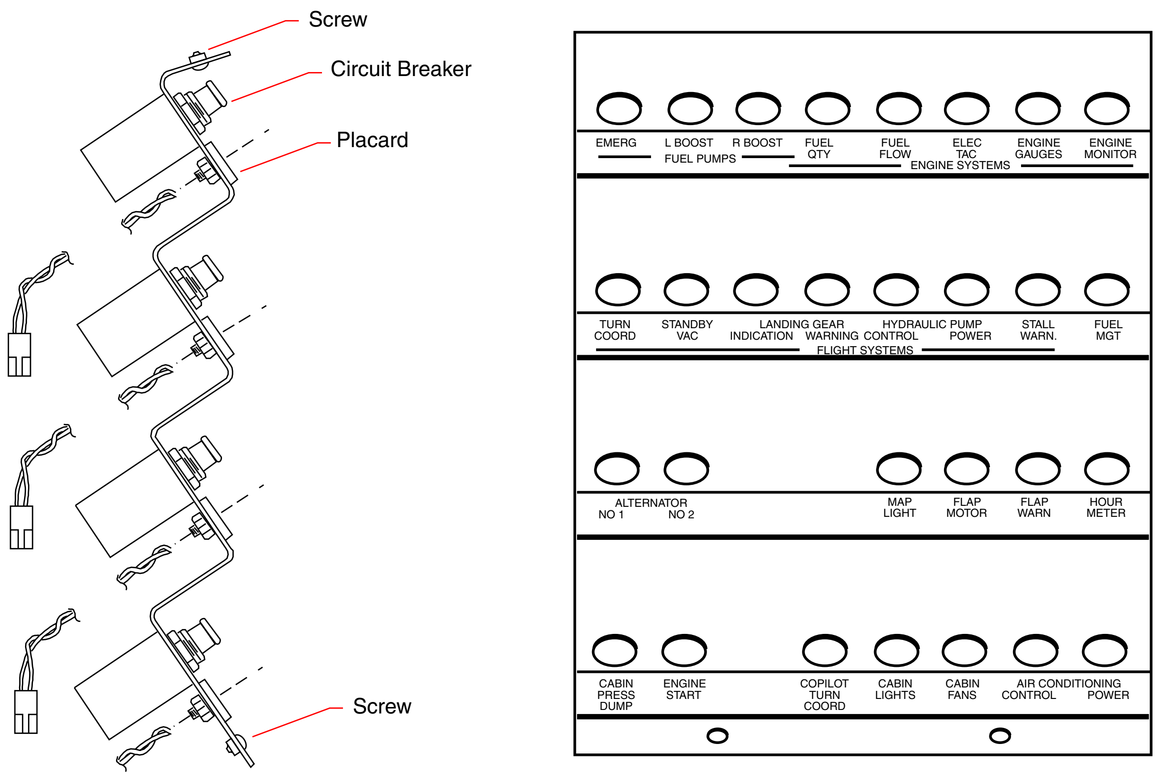
(f) Finish tightening jackscrew.

(g) Ensure that bezel is “tight” against panel.

2. Circuit Breaker Panels

See “Figure 3”.

In PA-46-350P (all aircraft) and PA-46R-350T with Garmin G1000 installed, circuit breakers are installed on both the pilot’s and copilot’s side. In PA-46R-350T with Avidyne installed, circuit breakers are installed on the lower right side of the instrument panel. Should a circuit breaker be replaced or added, exercise extreme caution ensuring the breakers are in proper mechanical alignment, any insulators that are called out are installed correctly, and all electrical wiring and connections meet aviation standards. Do not deviate from the parts manual requirements when replacing circuit breakers.



Profile View

Front View
(Typical Pilot's Fwd Panel Shown)

Circuit Breaker Panel - Typical - PA46-350P
Figure 3

3. Switches

NOTE: This airplane uses lighted switches which snap in from the front of the instrument panel and overhead switch panels.

In S/N's 4636001 thru 4636020, switches are of the rocker type and are mounted in the instrument panel.

In S/N's 4636021 thru 4636131, switches are of the rocker type and are mounted in the instrument panel and in overhead switch panels.

In S/N's 4636132 and up and S/N's 4692001 and up, most switches are of the rocker type and are mounted in the instrument panel and in overhead switch panels. The Environmental / Deice switch panel (see "Figure 4" on page 39109), however, is a sealed unit.

A. Overhead Switch Panel

When working on the overhead switch panel, remove it from the aircraft first.

(1) Removal

(a) Remove panel retaining screws (8).

(b) Slide the switch panel down to gain access to the electrical connectors at the back of the panel.

(c) Disconnect the electrical connectors and remove panel from aircraft.

(2) Installation

CAUTION: USE A MAXIMUM OF 10 ± 2 IN.-LBS. OF TORQUE WHEN INSTALLING THE SWITCH PANEL RETAINING SCREWS.

(a) Holding the switch panel below the opening in the headliner, connect the electrical connectors to the rear of the panel.

(b) Slide the panel into position and secure with screws (8). Torque screws to 10 ± 2 in.-lbs.

B. Rocker-type Electrical Switches

(1) Removal

(a) Gain access to the switch from behind the instrument panel or overhead switch panel. If switch is in an overhead switch panel, see Removal above.

(b) Squeeze retainer blades on top and bottom of the switch together and push switch from the panel.

(c) Make note of the placement of wires on the switch to facilitate installation.

(d) Disconnect wires from the switch. Remove switch.

(2) Installation

(a) Connect wires to switch.

(b) Push switch into the panel. Squeeze retainer blades on top and bottom of the switch together, if required.

(c) If switch is in an overhead switch panel, see Installation above.

C. Environmental / Deice Switch Panel

S/N's 4636132 and up and S/N's 4692001 and up

The Environmental / Deice Switch Panel in these airplanes is a front-mounted, sealed unit, which can not be repaired in the field.

(1) Removal

(a) Unscrew the four (4) #4 socket head cap screws and remove.

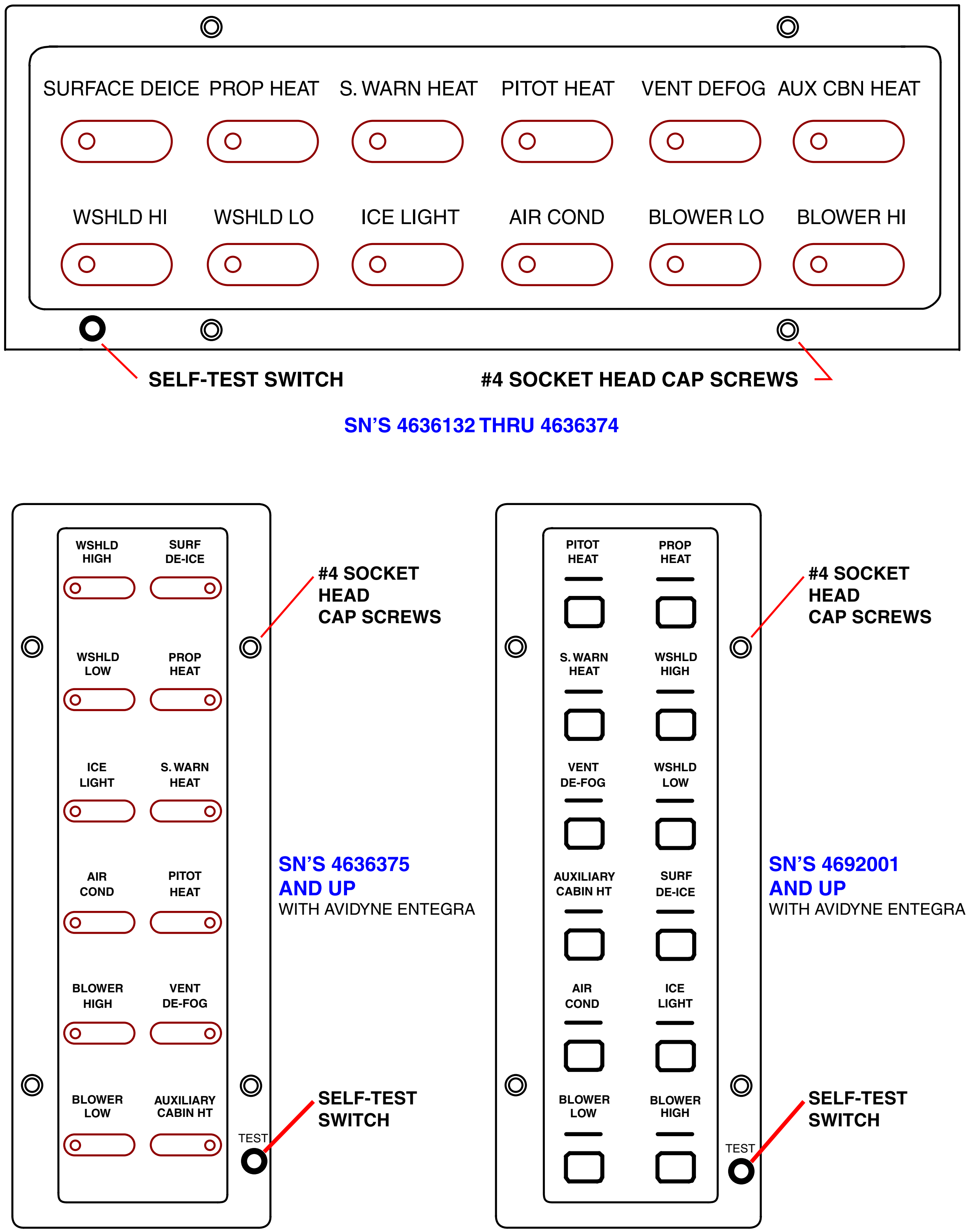
(b) Pull the unit out of the instrument panel enough to access the connectors attaching the unit to the wiring harness.

(c) Note arrangement of the connectors for re-installation. Disconnect the wiring harness and remove the unit.

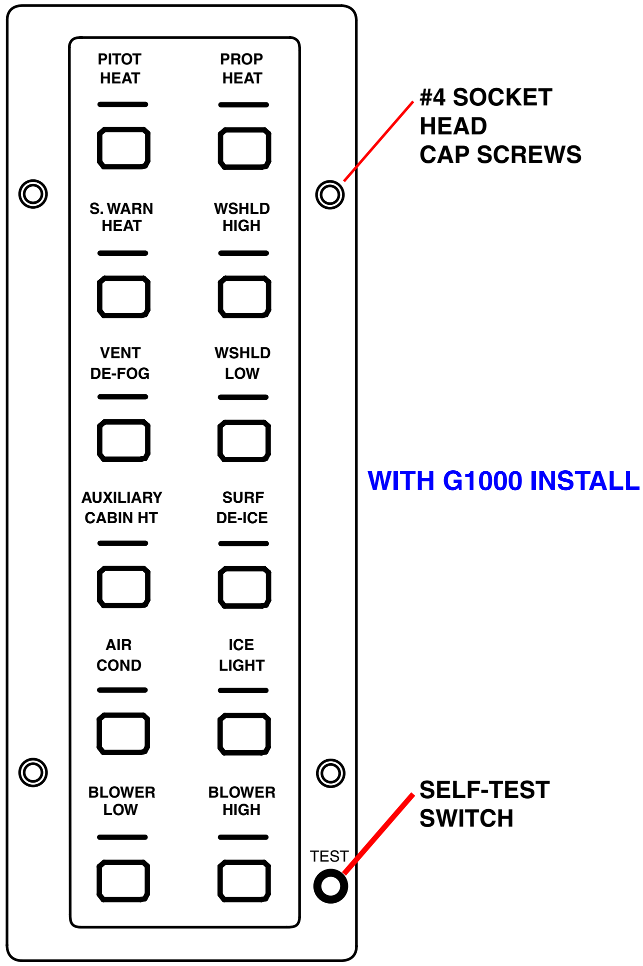
(2) Installation

(a) Connect the wiring harness and position the unit in the instrument panel.

(b) Secure with the four (4) #4 socket-head cap screws.



Environmental / Deice Switch Panel - Typical
Figure 4 (Sheet 1 of 2)



Environmental / Deice Switch Panel - Typical
Figure 4 (Sheet 2 of 2)

4. Linear Modular Connectors (LMD/LMS)

   Linear Modular Connectors (LMD/LMS) are used in some instrumentation and avionics installations to simplify assembly and reduce production costs. Special tools are needed to remove some modules in some installations. See Amphenol / Pyle publication LM-300 (LM-300-1) available from:

   http://www.amphenol-aerospace.com/ServiceInstructions.asp

5. EMI and RFI Check

   (PIR-PPS65119, Rev A.)

   A. Background

   ElectroMagnetic Interference (EMI) and Radio Frequency Interference (RFI) are a growing concern with the installation of advanced digital avionics. The following test procedure is recommended before return-to-service each time any electrical / avionics work is performed.

   B. Procedure

   (1) Position the aircraft at least 50 yards from buildings or any other large structures.

   (2) Ensure all aircraft closeout panels (excluding interior trim panels) and engine cowling are installed on the aircraft being tested.

   (3) With the aircraft running and all avionics, exterior lights and equipment ON, verify that:

   (a) There are no unusual needle or display fluctuations on any display or gauge;

   (b) There is no objectionable background noise in the headsets.

   (c) For troubleshooting purposes, if either is present, systematically turn OFF equipment until the offending system is identified.

   (4) RFI interference is typically generated by energy from the aircraft communication radios bouncing around the airframe. Verify that transmitting on the communication system does not cause RFI interference by performing the following test on each communication system.

   (a) On each of the following frequencies key the microphone for 3 to 5 seconds:

121.150 MHZ	127.000 MHZ
131.250 MHZ	121.175 MHZ
121.200 MHZ	131.275 MHZ
123.000 MHZ	131.300 MHZ

   (b) Verify that this does not cause any unusual needle or display fluctuations on any display or gauge.

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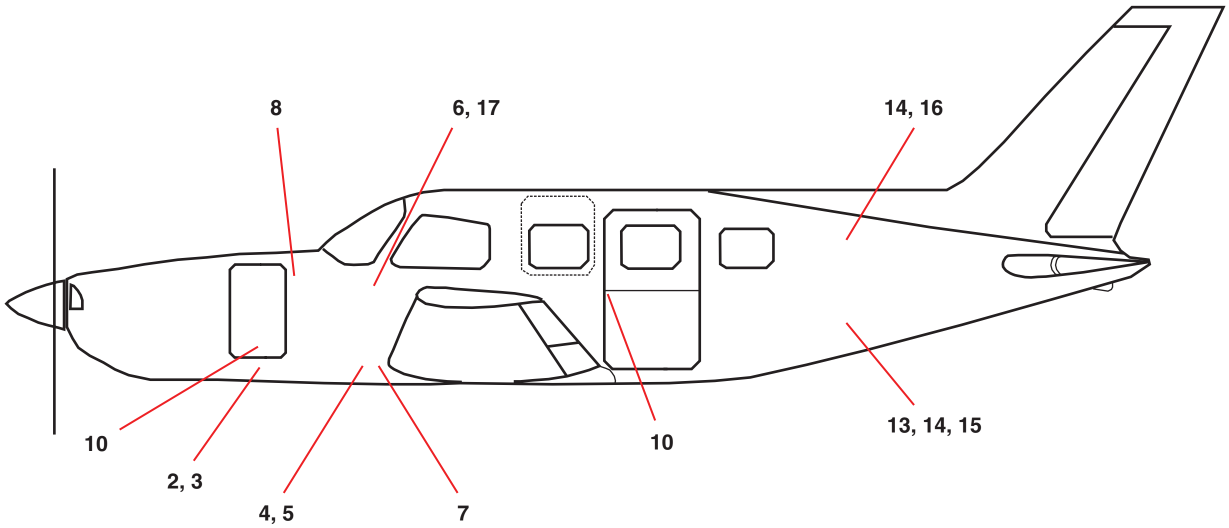
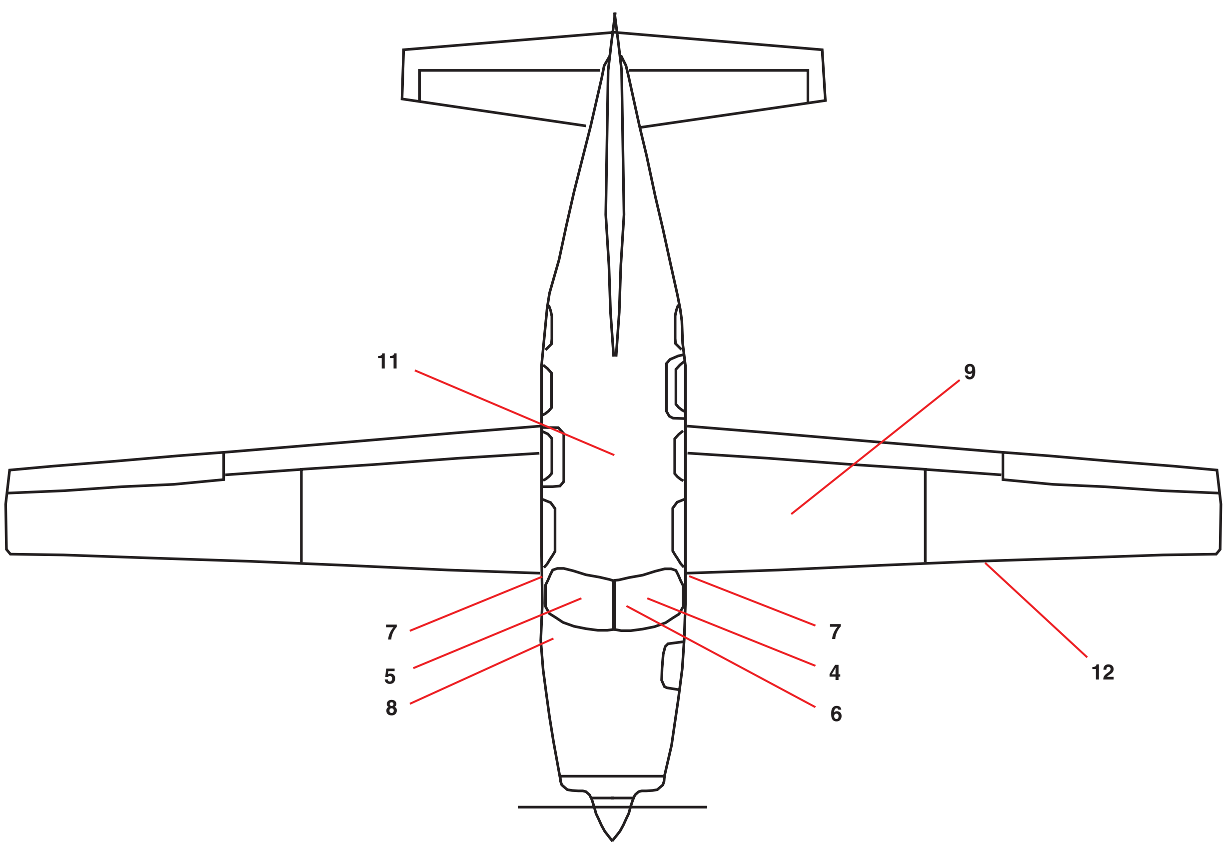
ELECTRICAL AND ELECTRONIC EQUIPMENT

The following illustrations depict the location and installation of various electrical/electronic components. “Figure 1” is an orientation view showing the airframe locations covered by the following figures.

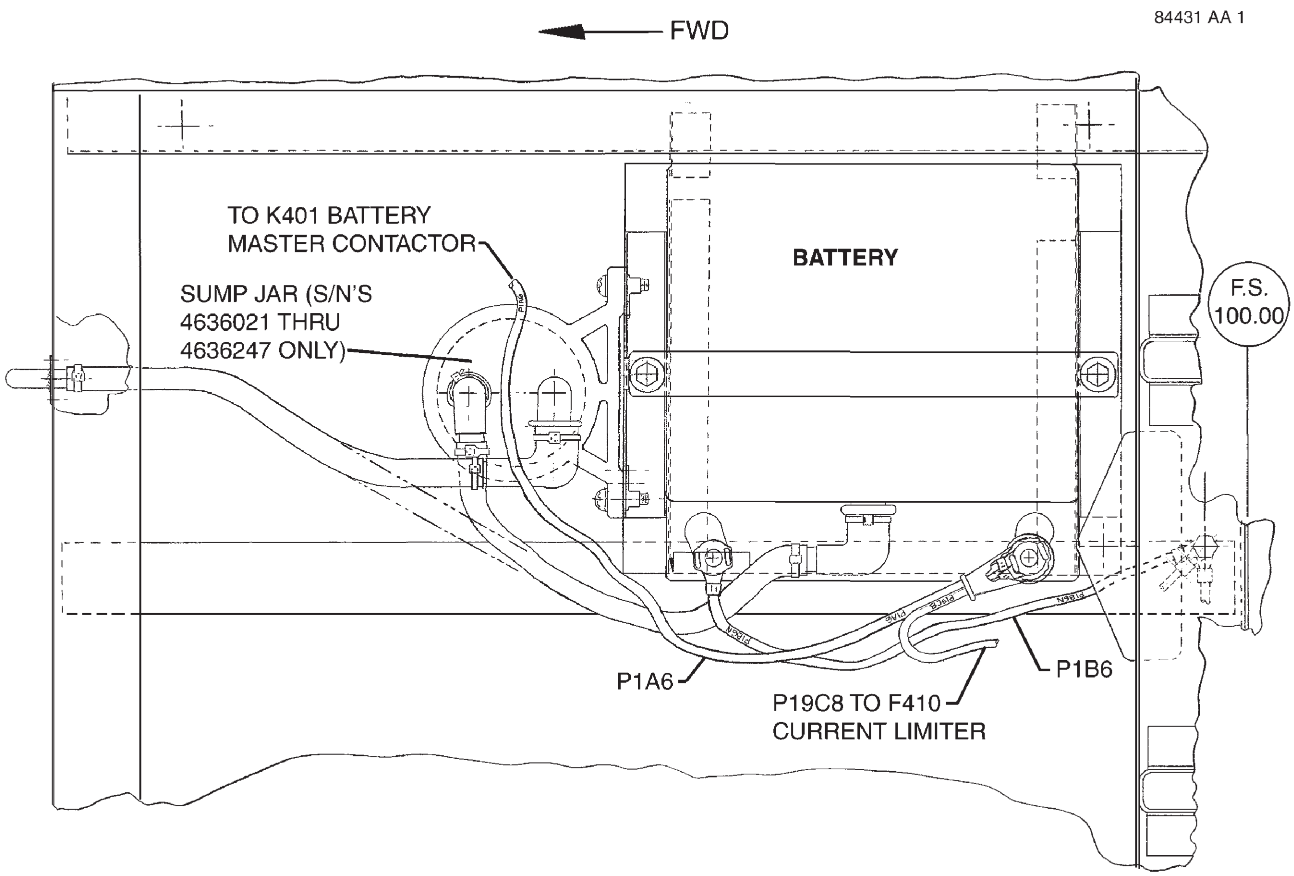
Index



ITEM NUMBERS KEYED TO FIGURE NUMBER

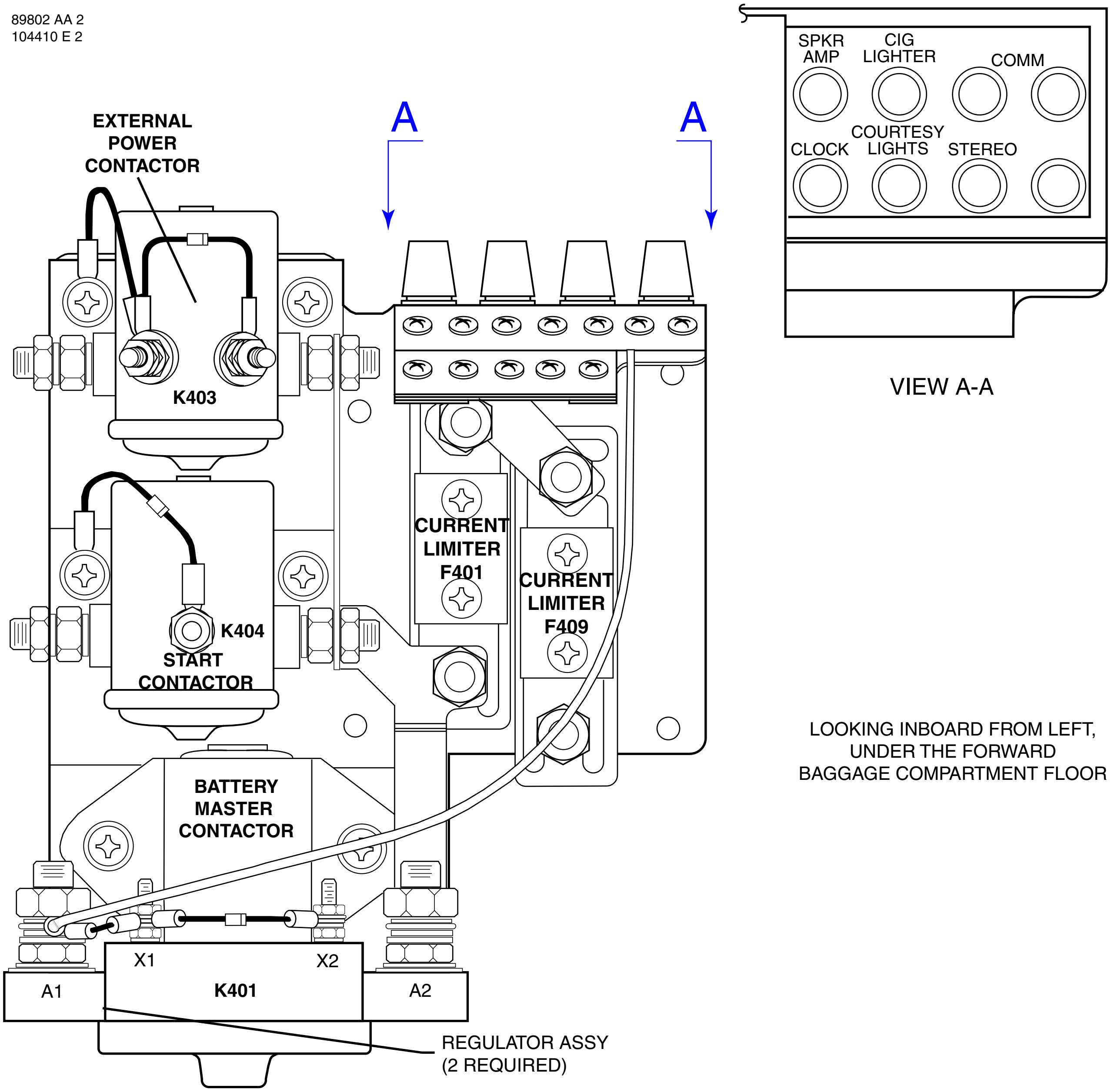


Electrical/Electronic Component Locator
Figure 1



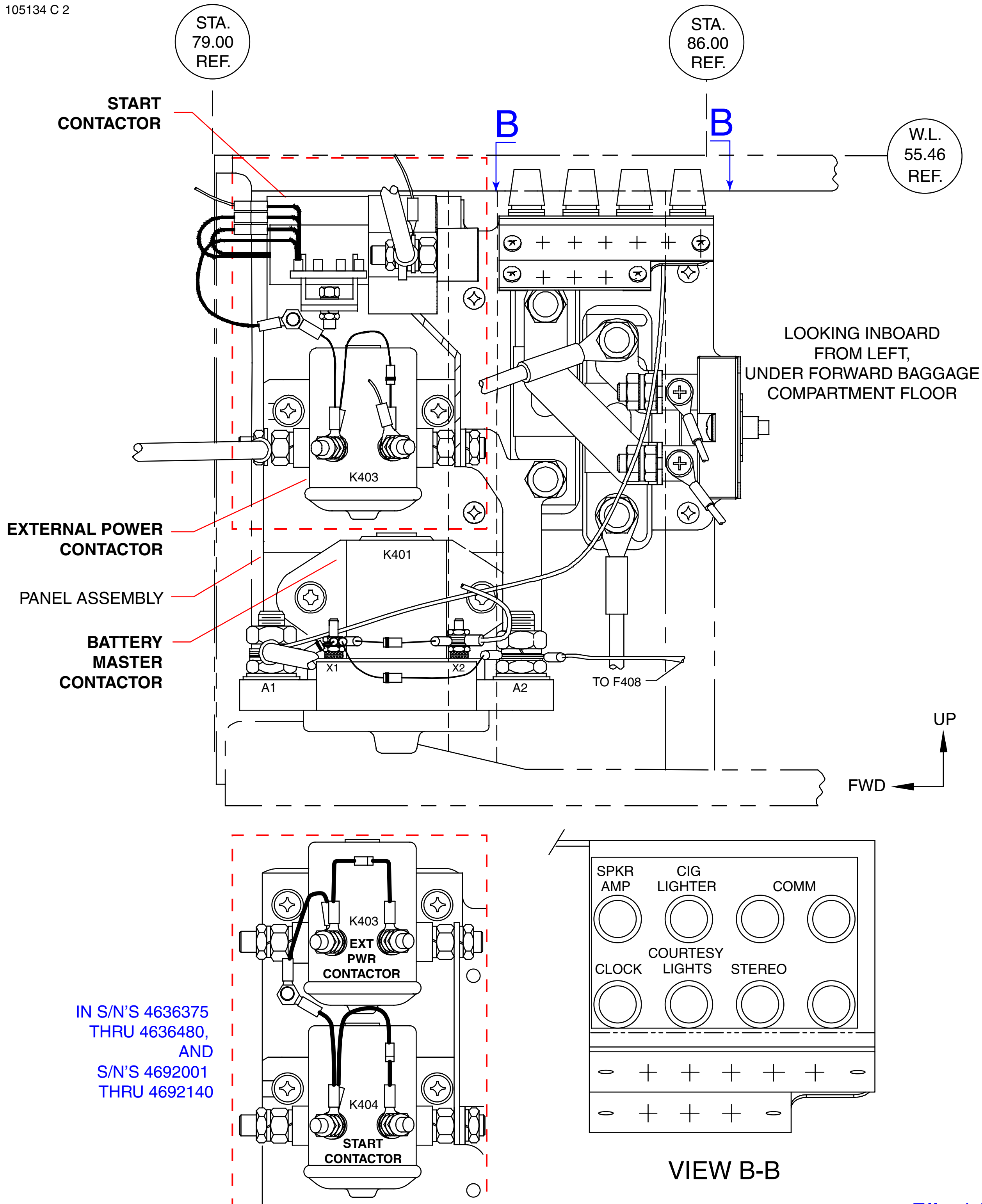
LEFT SIDE, LOOKING DOWN, BAGGAGE COMPARTMENT FLOOR REMOVED

Electrical/Electronic Component Locator
Figure 2



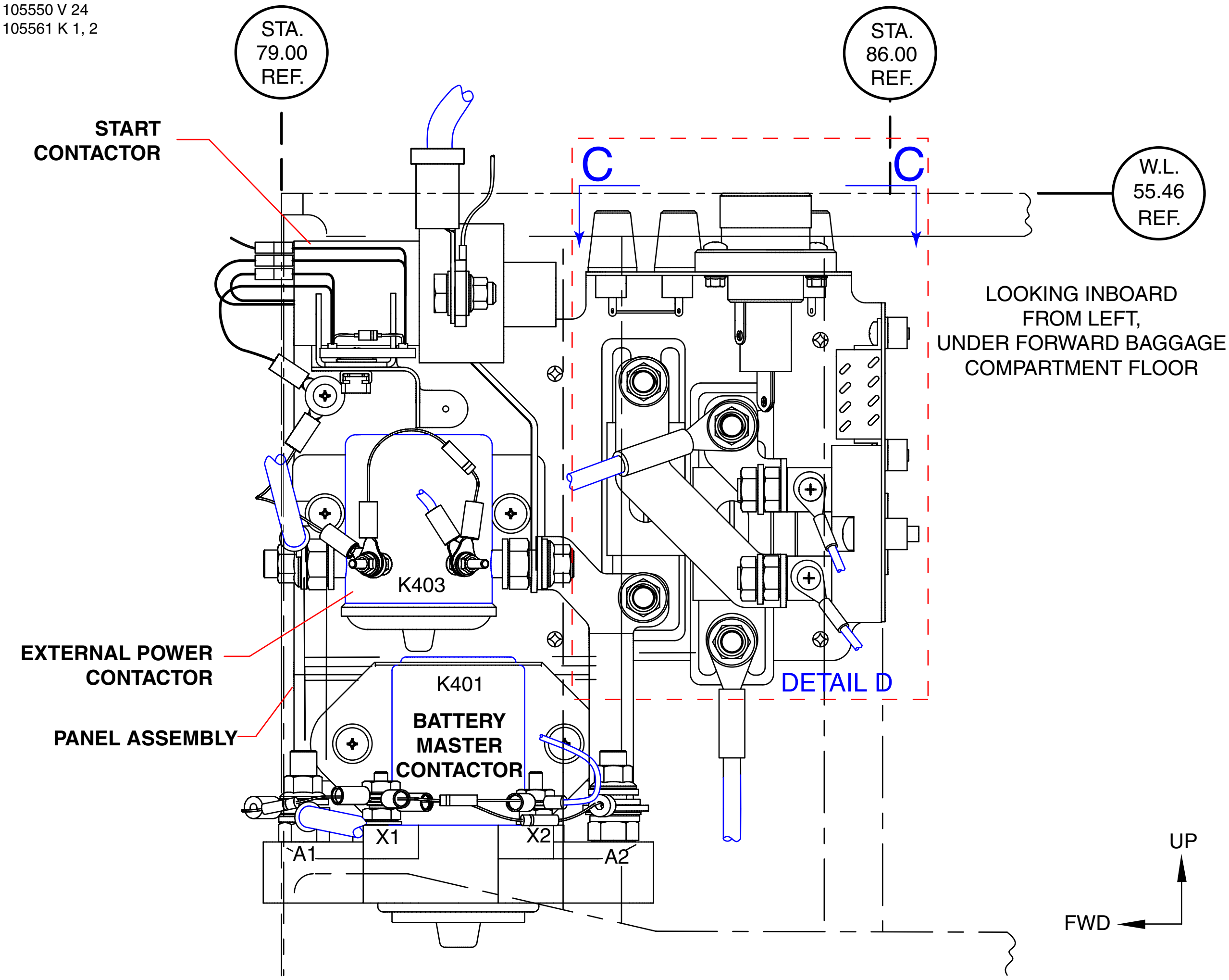
Electrical/Electronic Component Locator
Figure 3 (Sheet 1 of 3)

Effectivity
4636001 thru 4636374

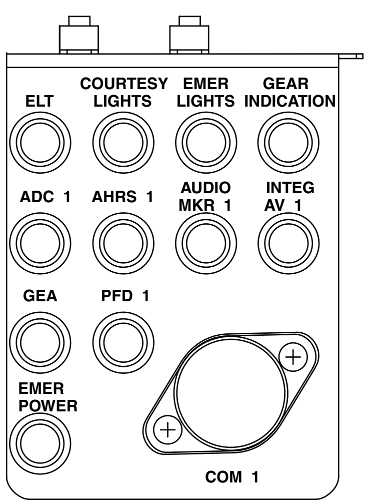


Effectivity
4636375–4636459
4636461–4636462, 4636481
4692001–4692133, 4692141
4692149 and 4692153

Electrical/Electronic Component Locator
Figure 3 (Sheet 2 of 3)

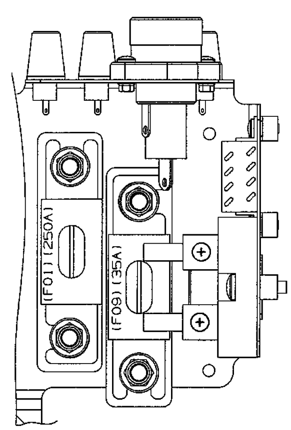


VIEW C-C



(TYPICAL)

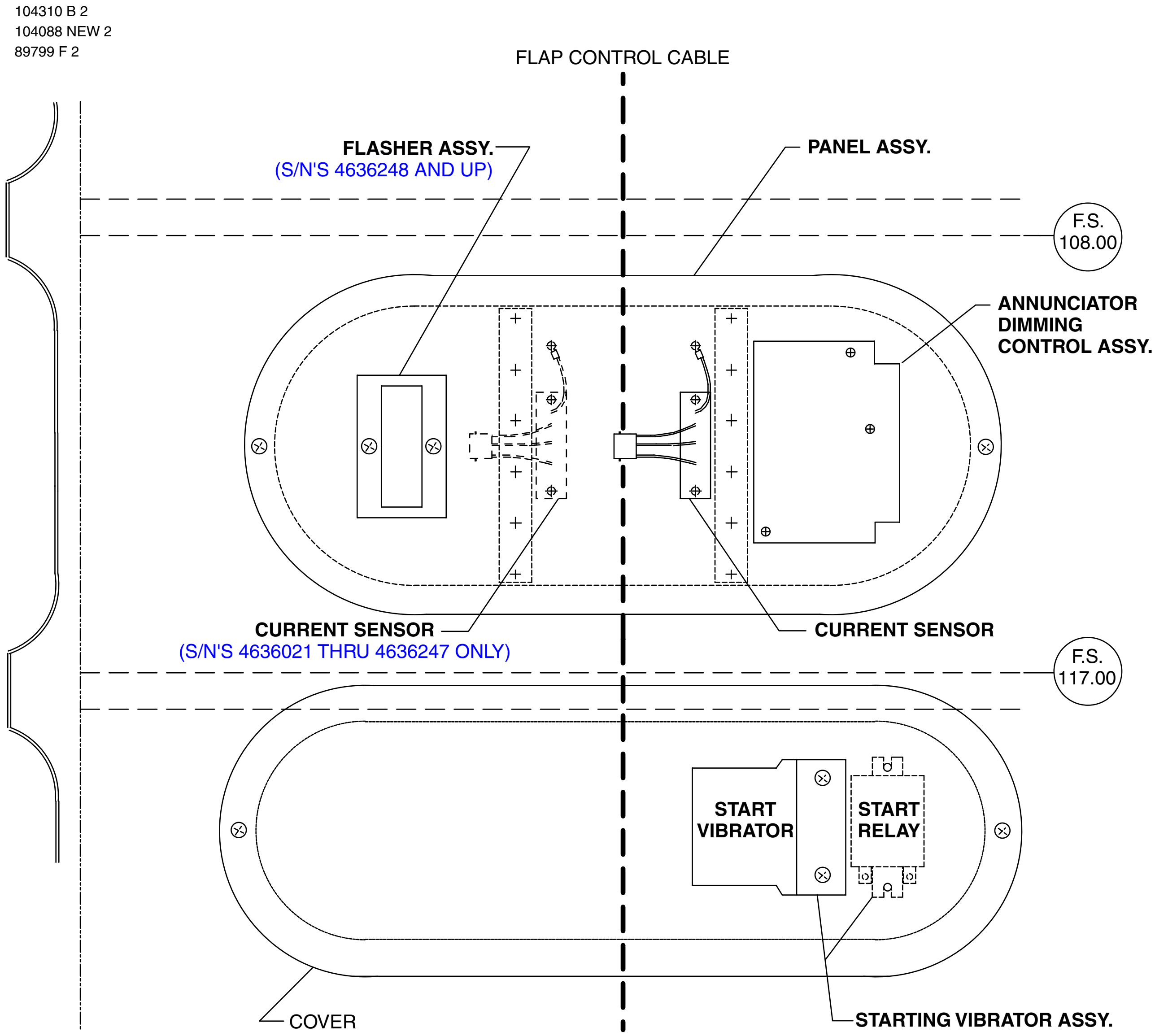
DETAIL D



Effectivity
with Garmin G1000

Electrical/Electronic Component Locator
Figure 3 (Sheet 3 of 3)

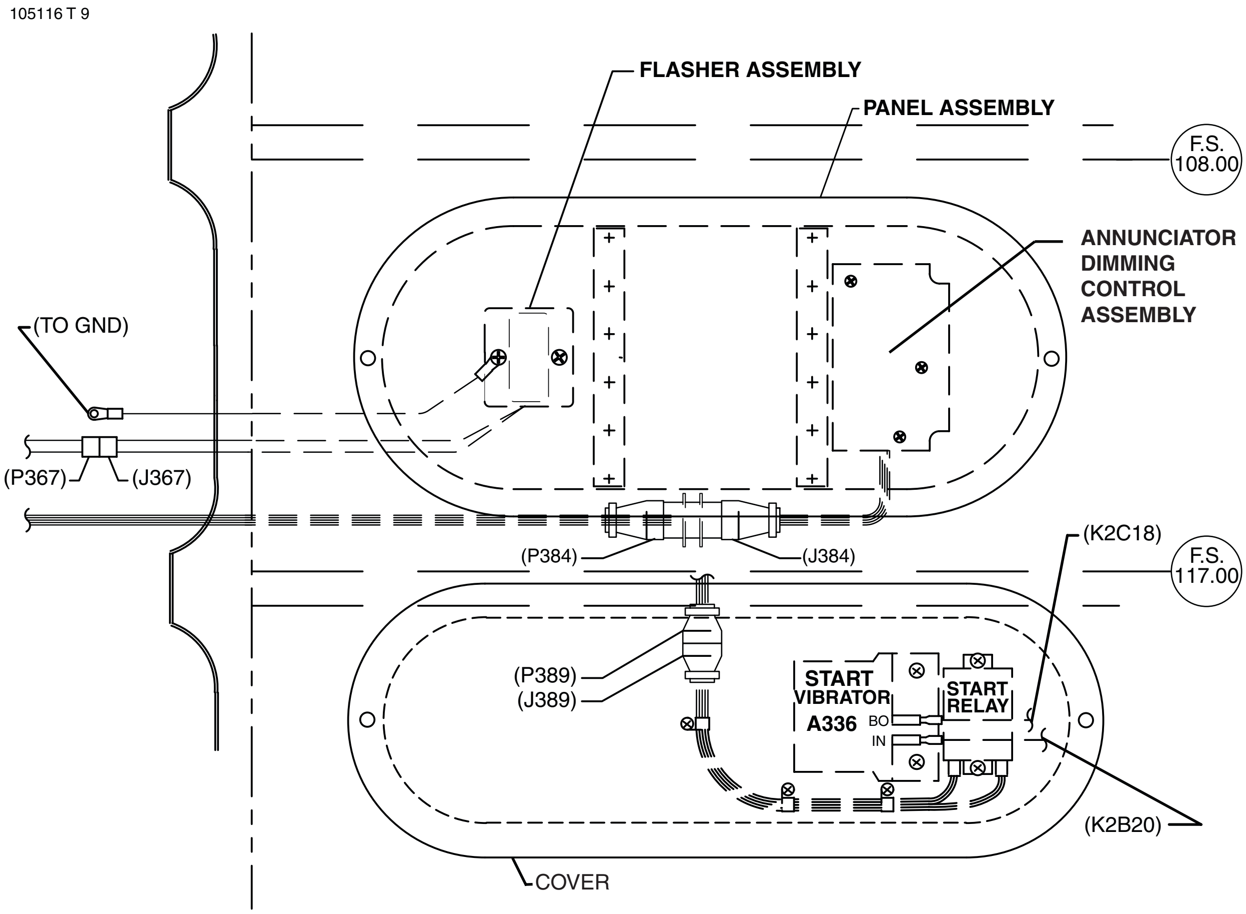
39206



RELAY INSTALLATION - PILOT SIDE

Electrical/Electronic Component Locator
Figure 4 (Sheet 1 of 4)

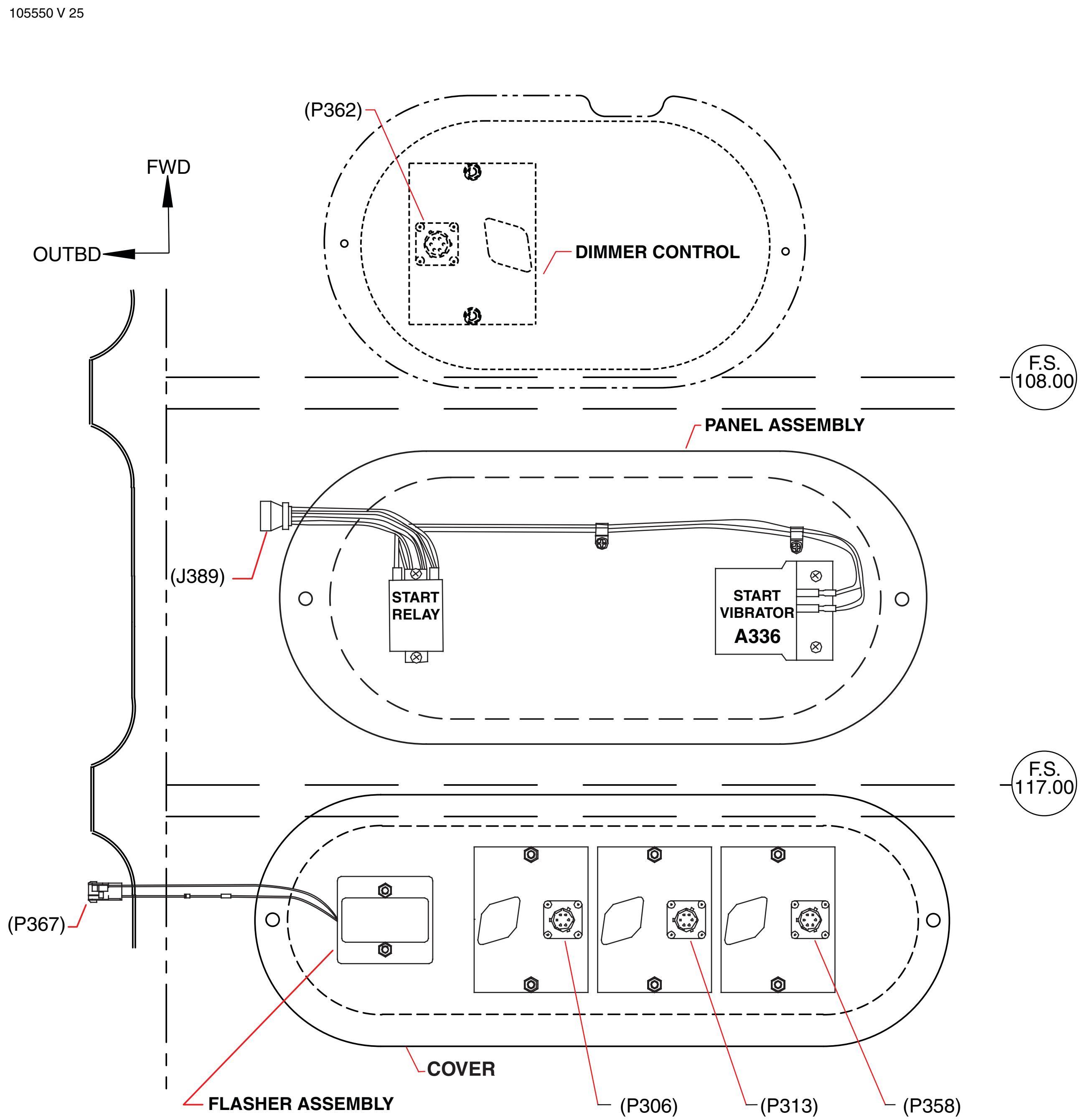
Effectivity
4636001 thru 4636374



RELAY INSTALLATION - PILOT SIDE

Effectivity
4636375–4636459
4636461–4636462, 4636481
4692001–4692133, 4692141
4692149 and 4692153

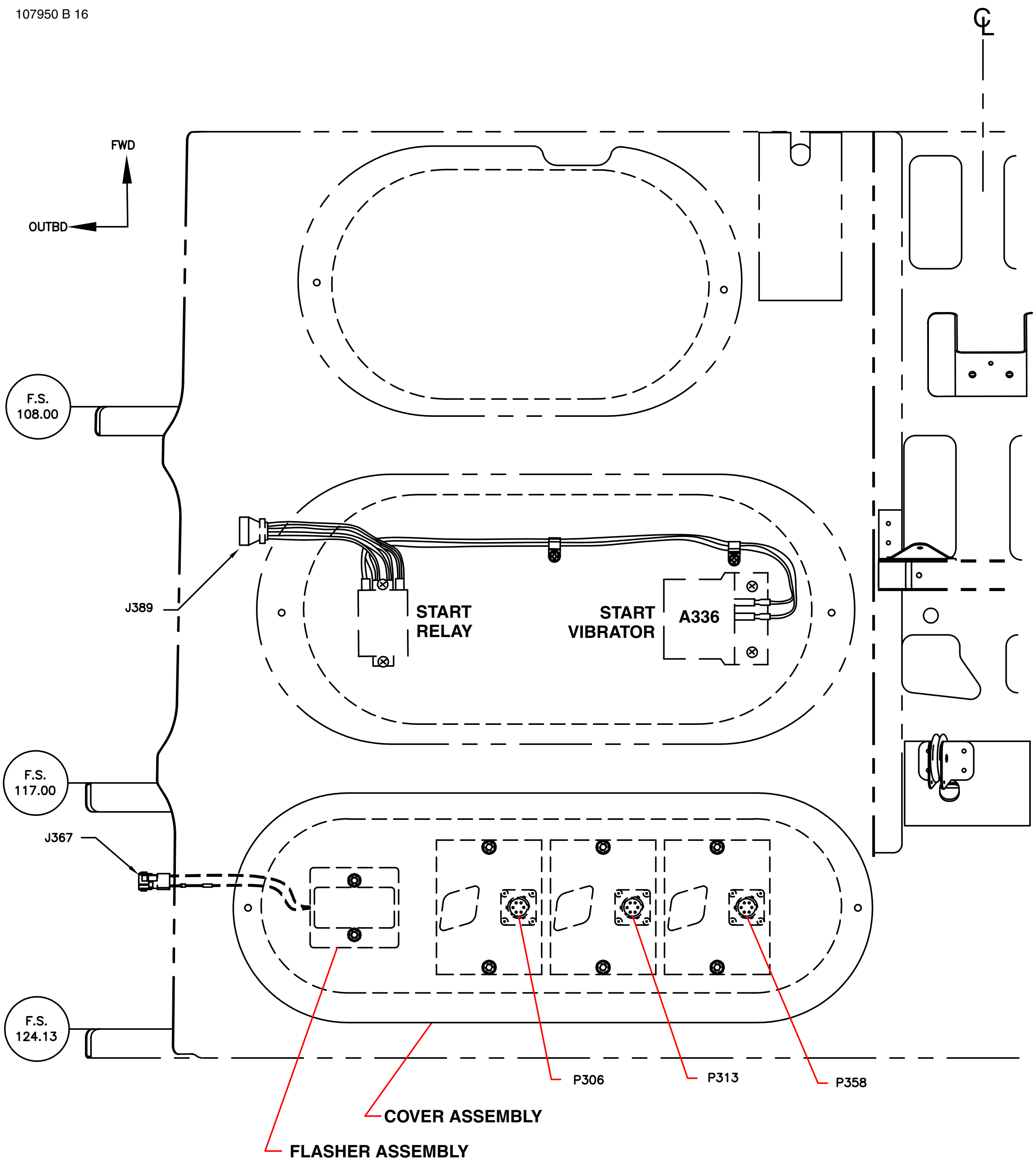
Electrical/Electronic Component Locator
Figure 4 (Sheet 2 of 4)



RELAY INSTALLATION - PILOT SIDE

Electrical/Electronic Component Locator
Figure 4 (Sheet 3 of 4)

Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153

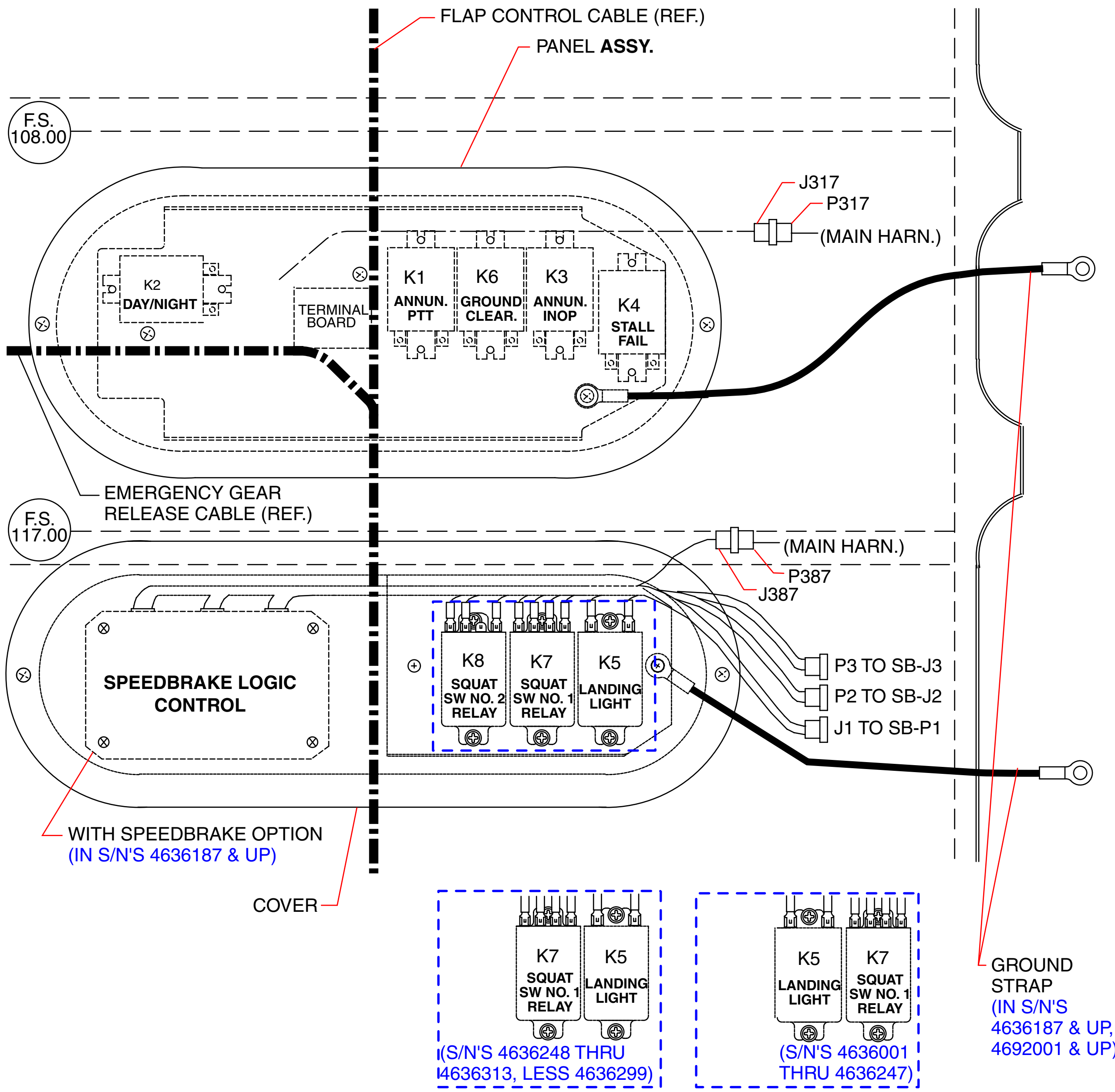


RELAY INSTALLATION - PILOT SIDE
(LOOKING DOWN)

Effectivity
4636633, 4636652 and up

Electrical/Electronic Component Locator
Figure 4 (Sheet 4 of 4)

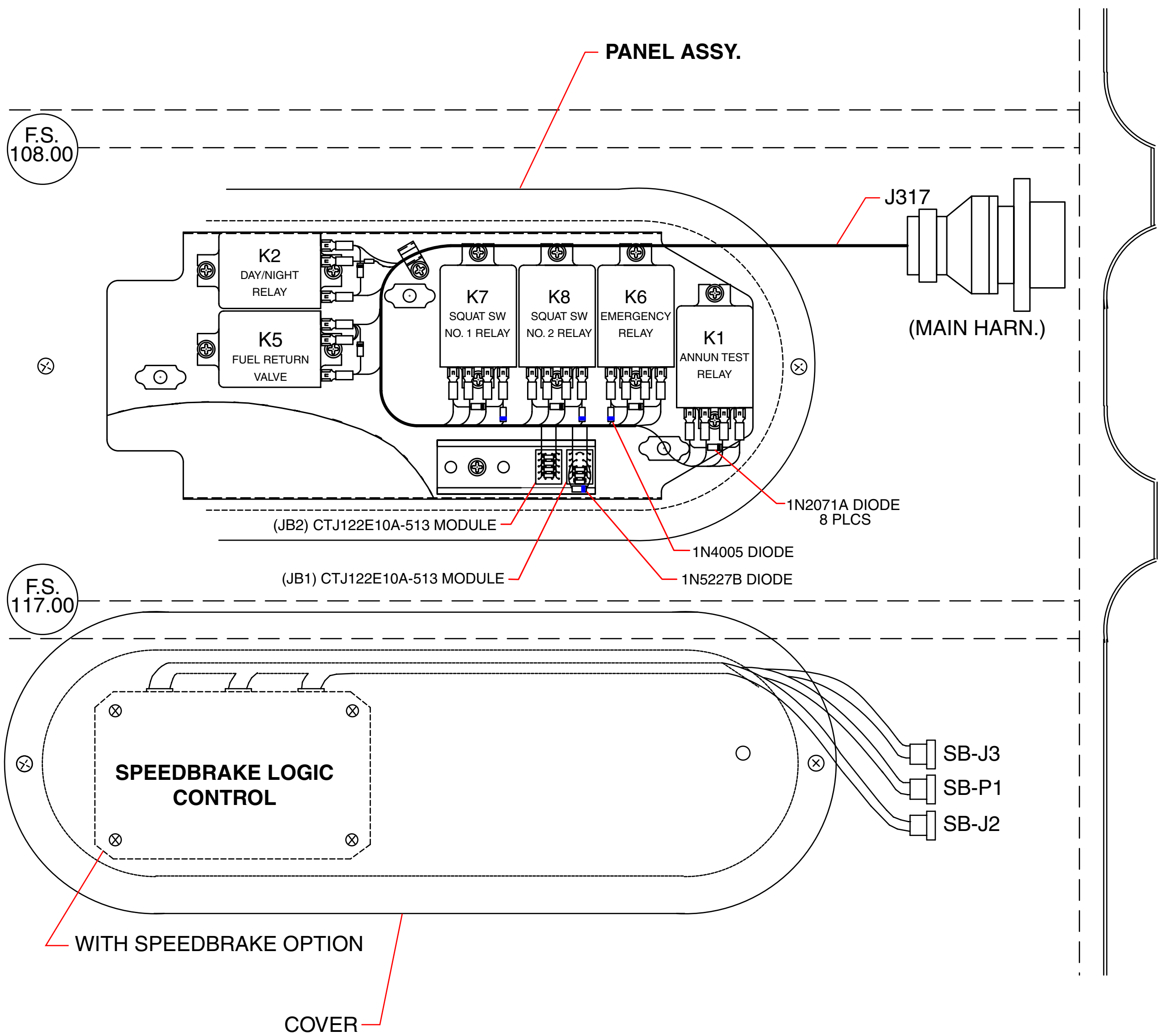
104310 B 1
104088 NEW 1
89799 F 1



RELAY INSTALLATION - CO-PILOT SIDE

Electrical/Electronic Component Locator
Figure 5 (Sheet 1 of 4)

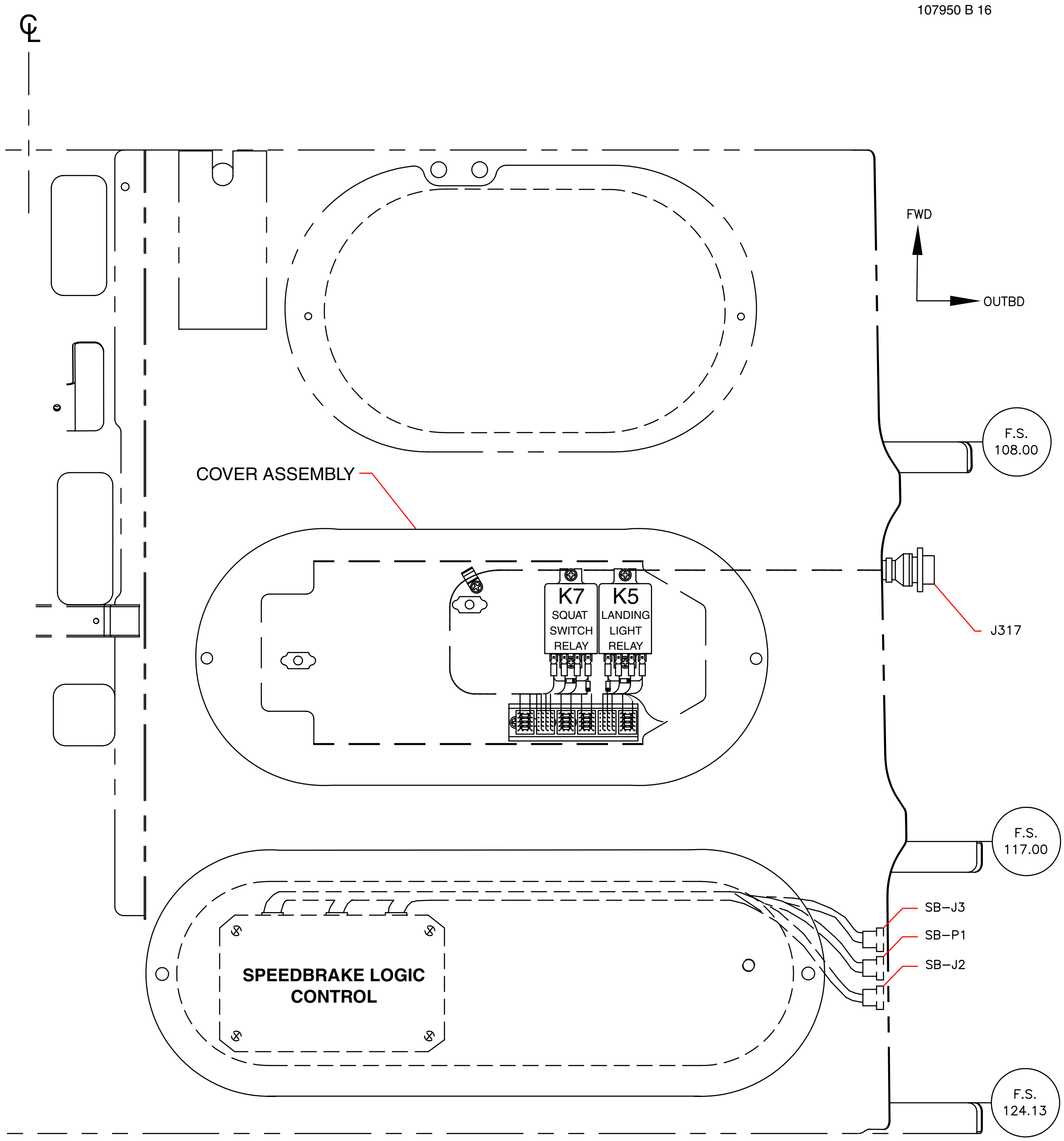
105550 V 25



RELAY INSTALLATION - CO-PILOT SIDE

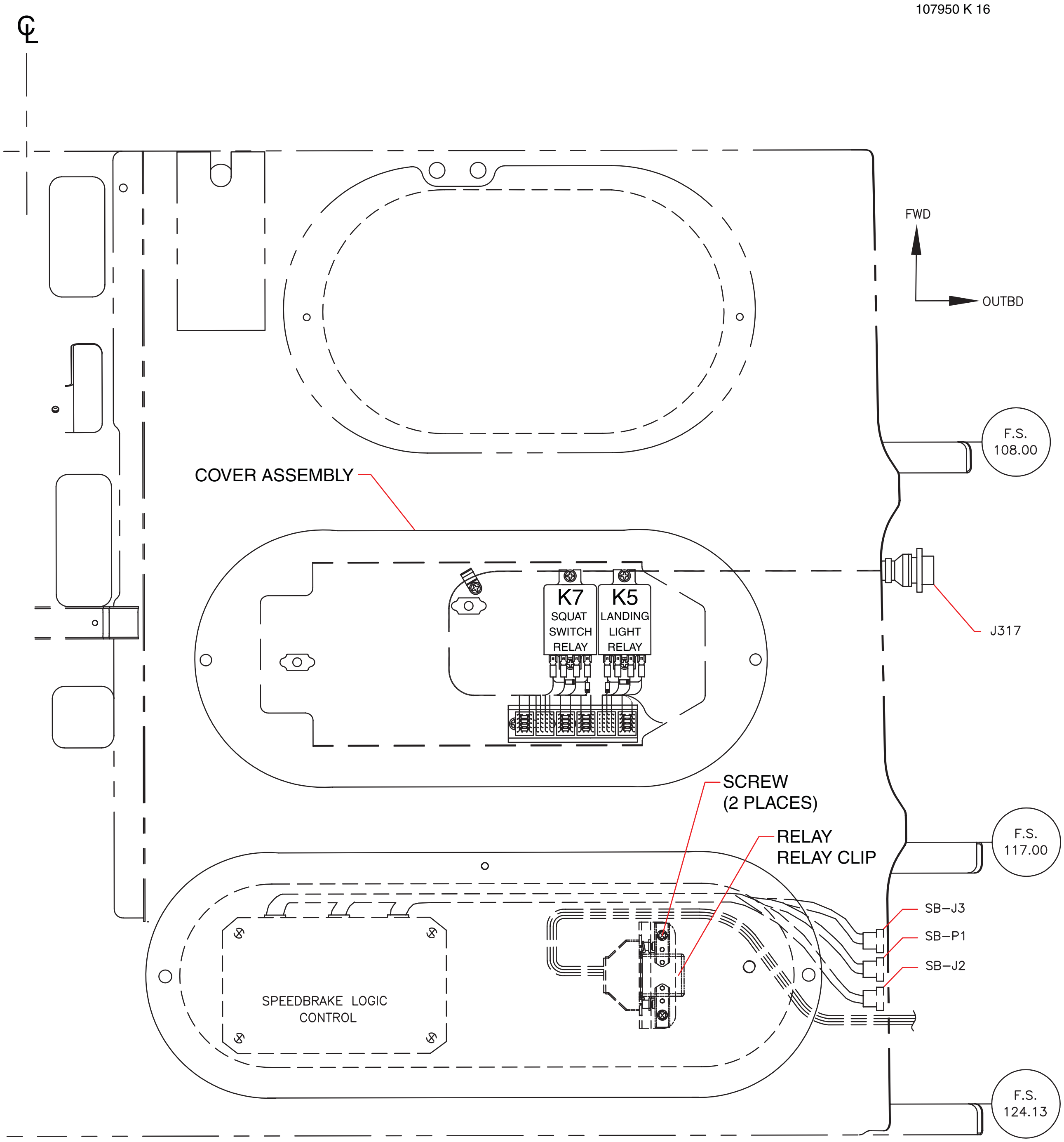
Effectivity
4636460, 4636463–4636651
less 4636481 and 4636633
4692134 and up, less 4692141
4692149 and 4692153

Electrical/Electronic Component Locator
Figure 5 (Sheet 2 of 4)



RELAY INSTALLATION - CO-PILOT SIDE
(LOOKING DOWN)

Electrical/Electronic Component Locator
Figure 5 (Sheet 3 of 4)



RELAY INSTALLATION - CO-PILOT SIDE
(LOOKING DOWN)

Effectivity
4636697 and up; and,
4636633, 4636652–4636696
when Piper Kit No. 88589-002
is installed per latest revision
of SL 1201

Electrical/Electronic Component Locator
Figure 5 (Sheet 4 of 4)

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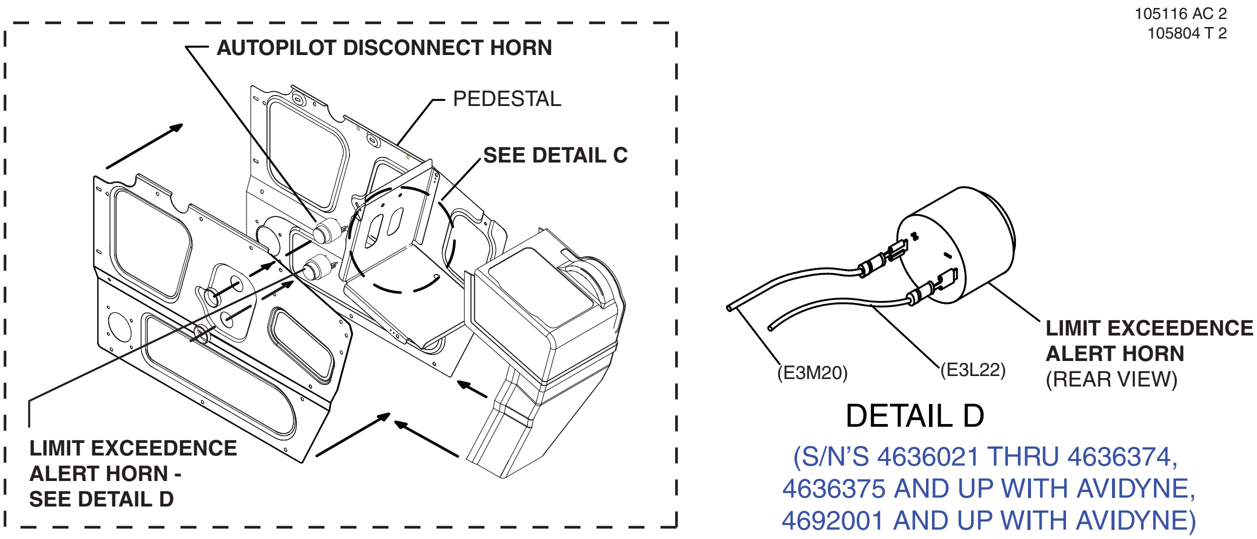
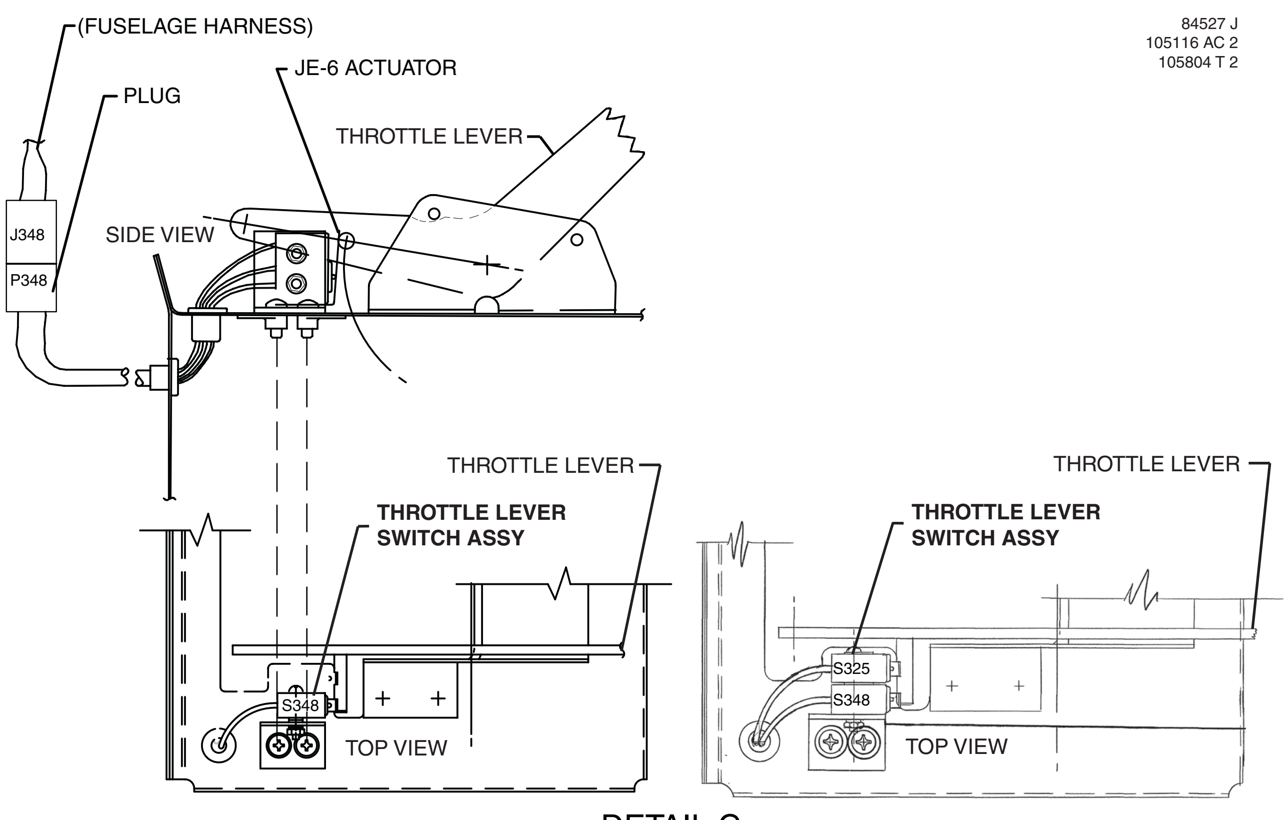


DETAIL D

(S/N'S 4636021 THRU 4636374, 4636375 AND UP WITH AVIDYNE, 4692001 AND UP WITH AVIDYNE)

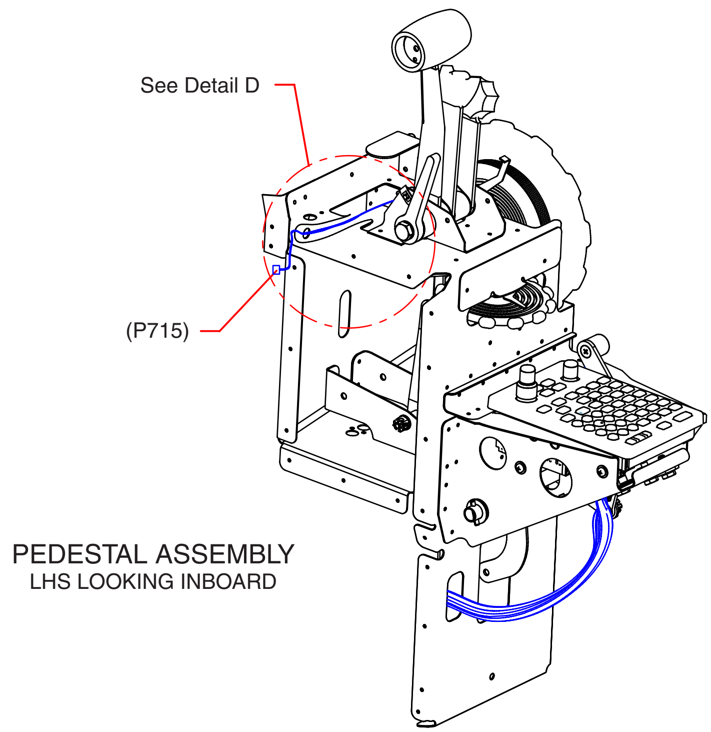


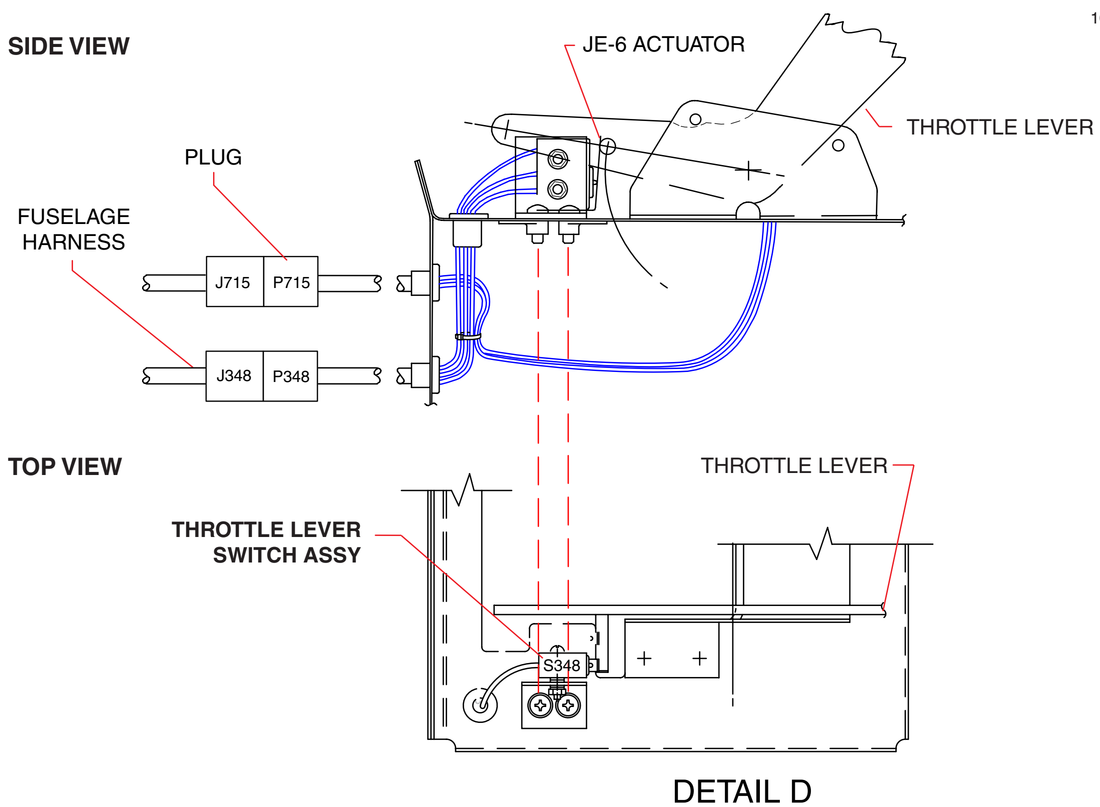
DETAIL C

(S/N'S 4636375 AND UP, 4692001 AND UP)

(S/N'S 4636001 THRU 4636374)

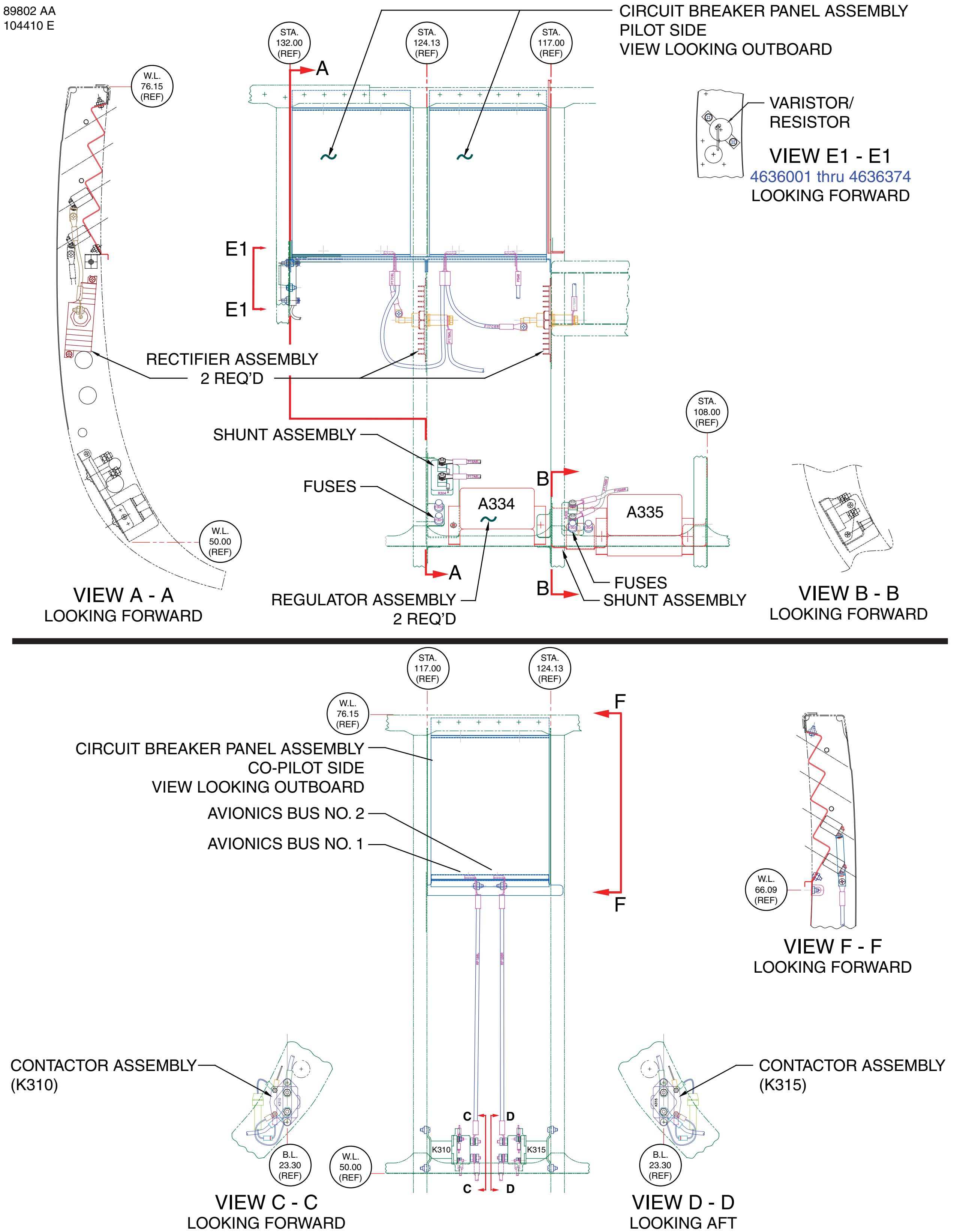
Electrical/Electronic Component Locator
Figure 6 (Sheet 1 of 2)





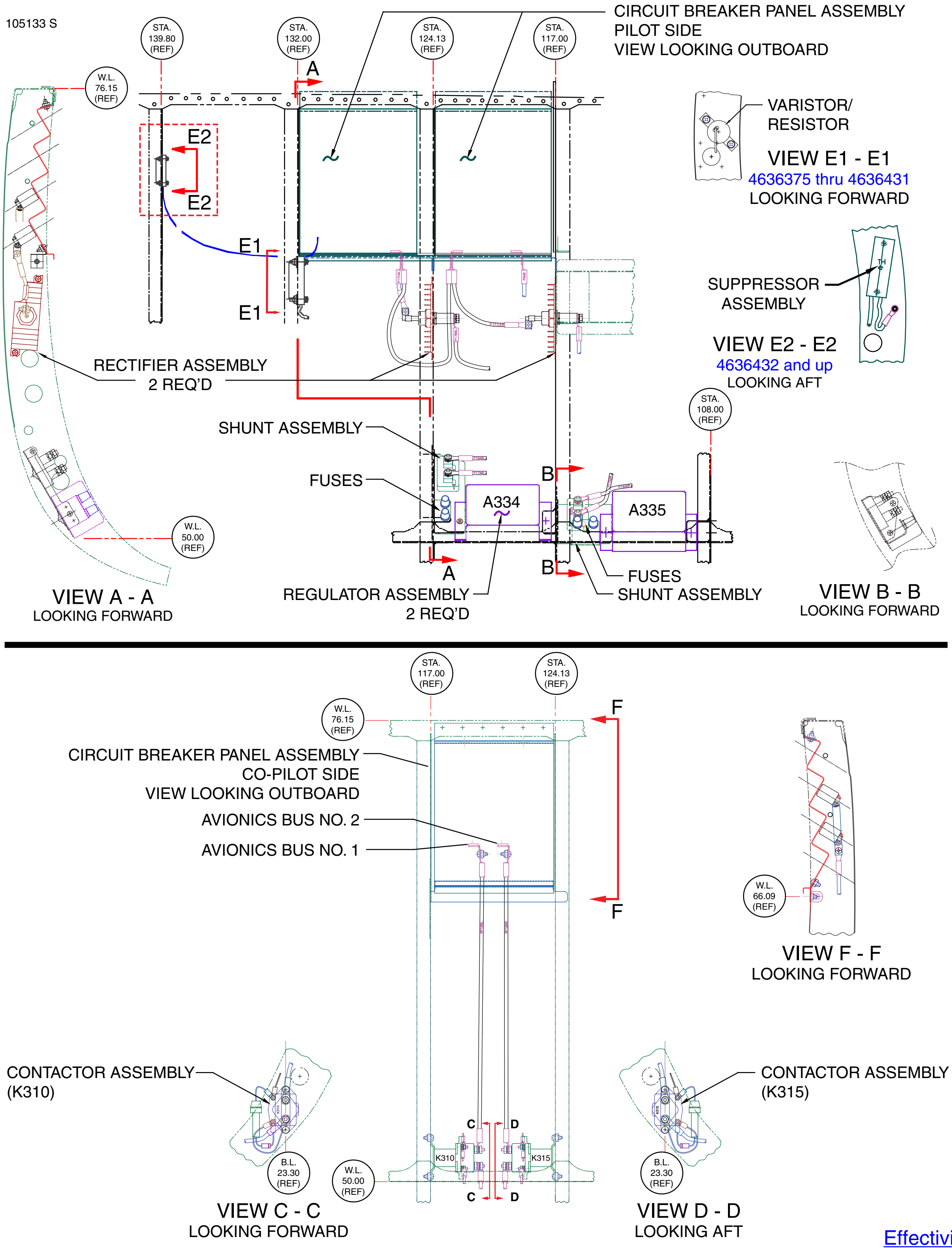
Electrical/Electronic Component Locator
Figure 6 (Sheet 2 of 2)

Effectivity
4636460, 4636463 and up,
4692134 and up



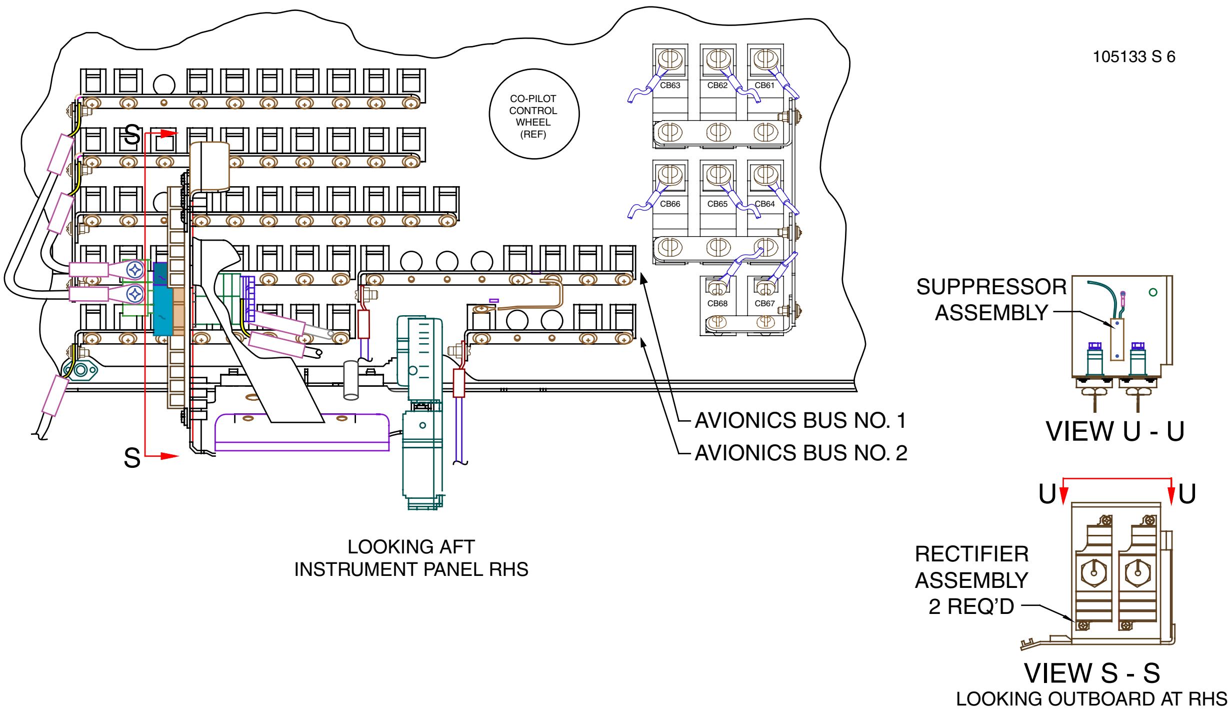
Effectivity
4636001 thru 4636374

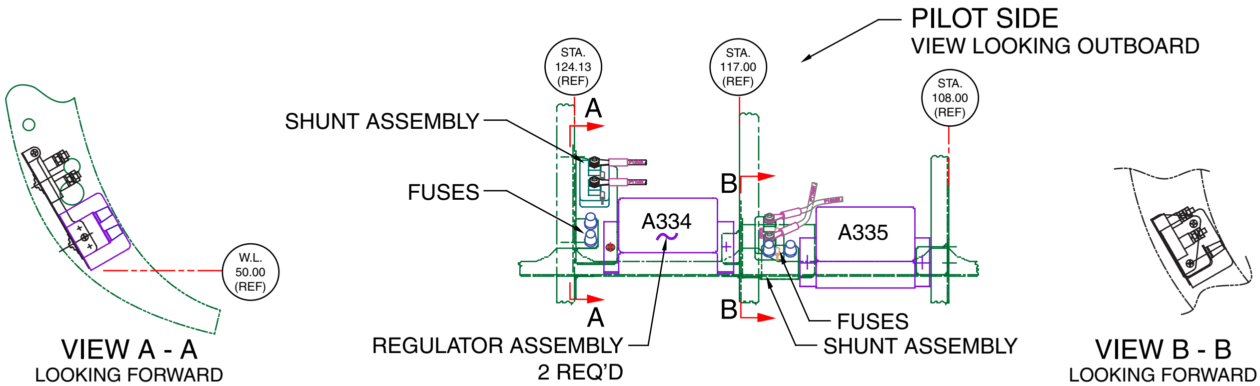
Electrical/Electronic Component Locator
Figure 7 (Sheet 1 of 4)

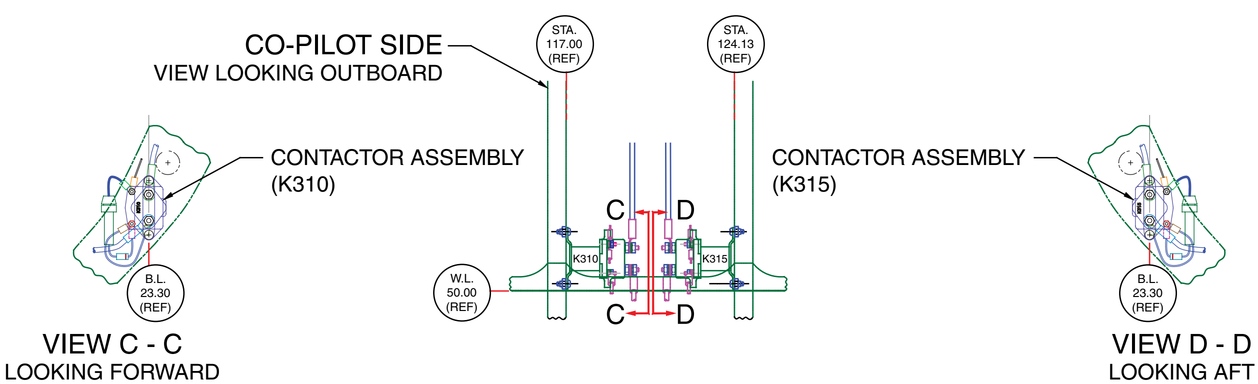


Effectivity
4636375–4636459
4636461–4636462, 4636481

Electrical/Electronic Component Locator
Figure 7 (Sheet 2 of 4)

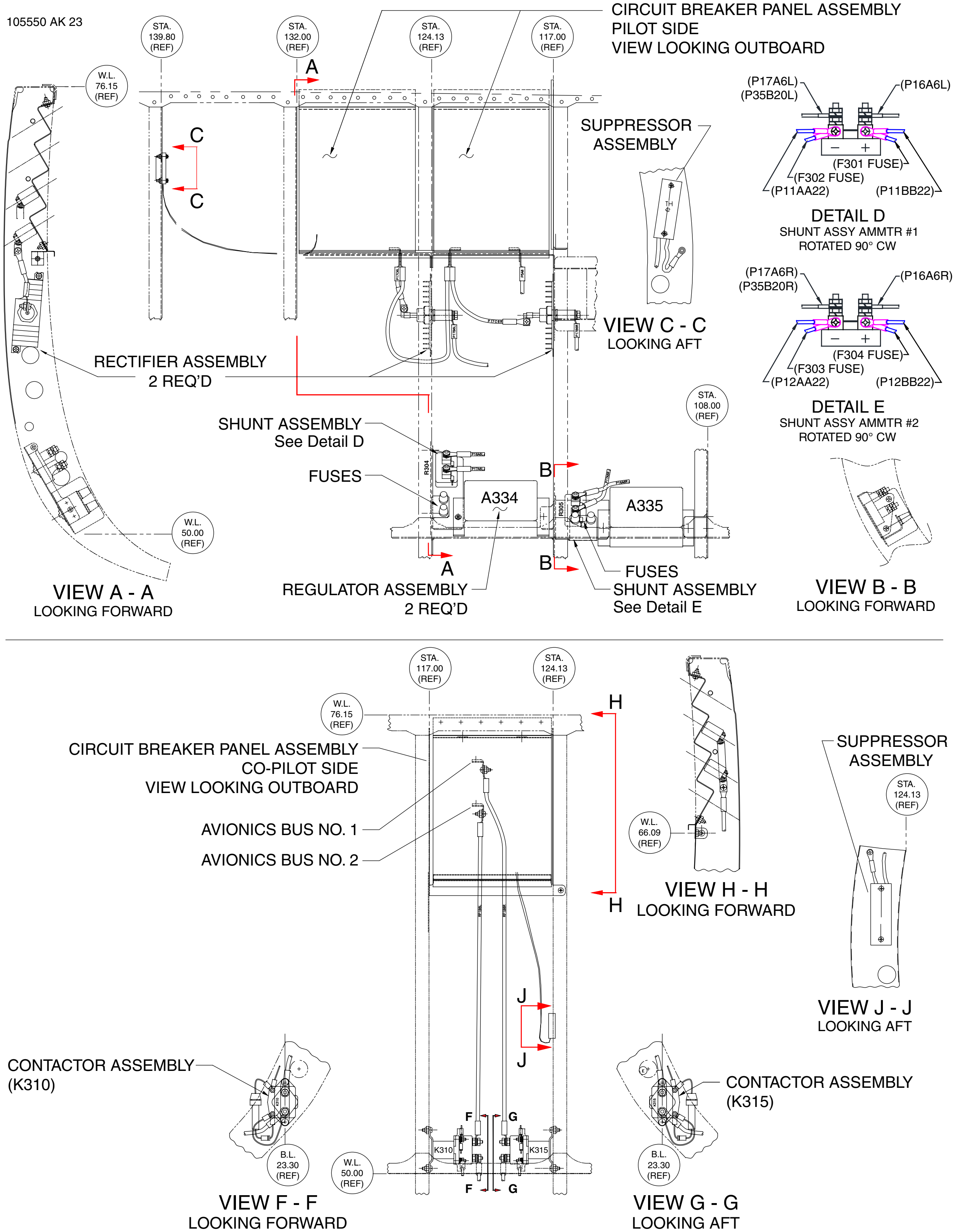






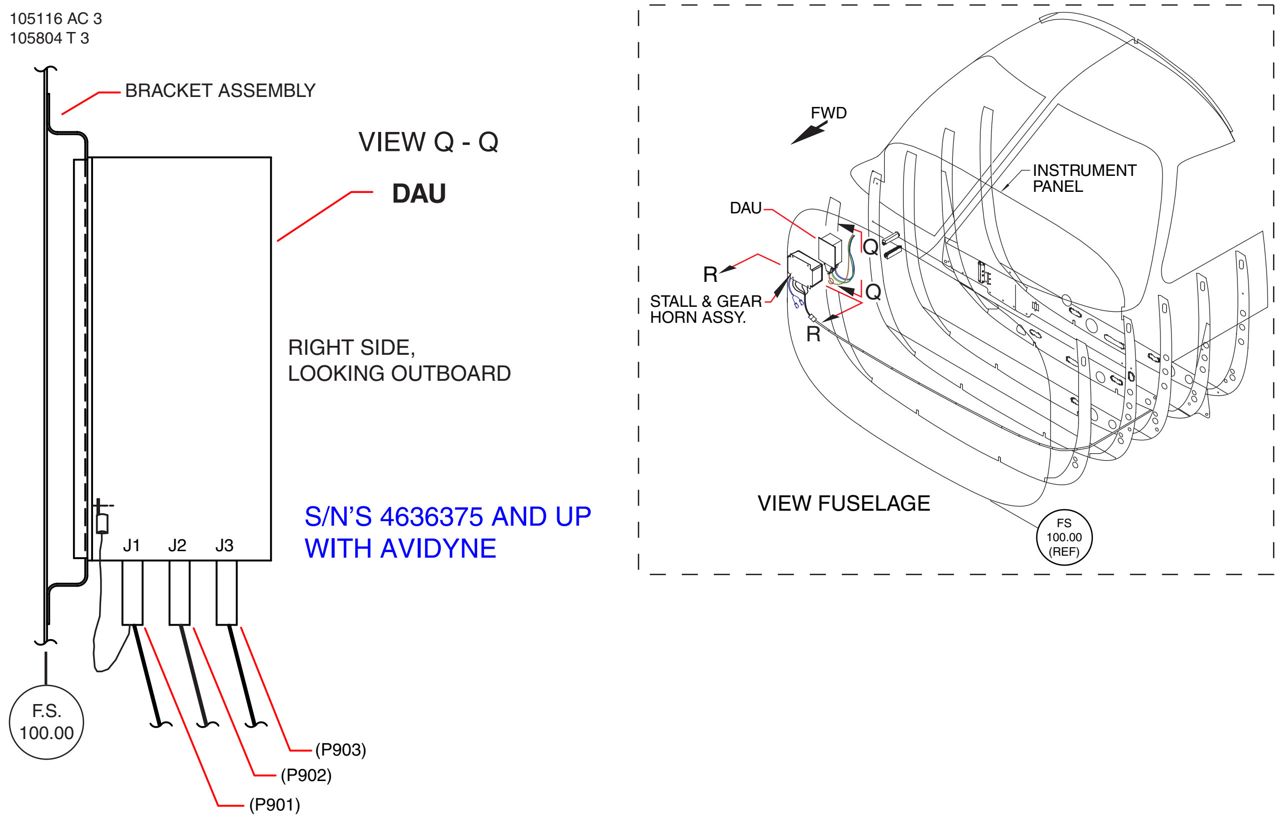
Effectivity
4692001–4692133, 4692141
4692149 and 4692153

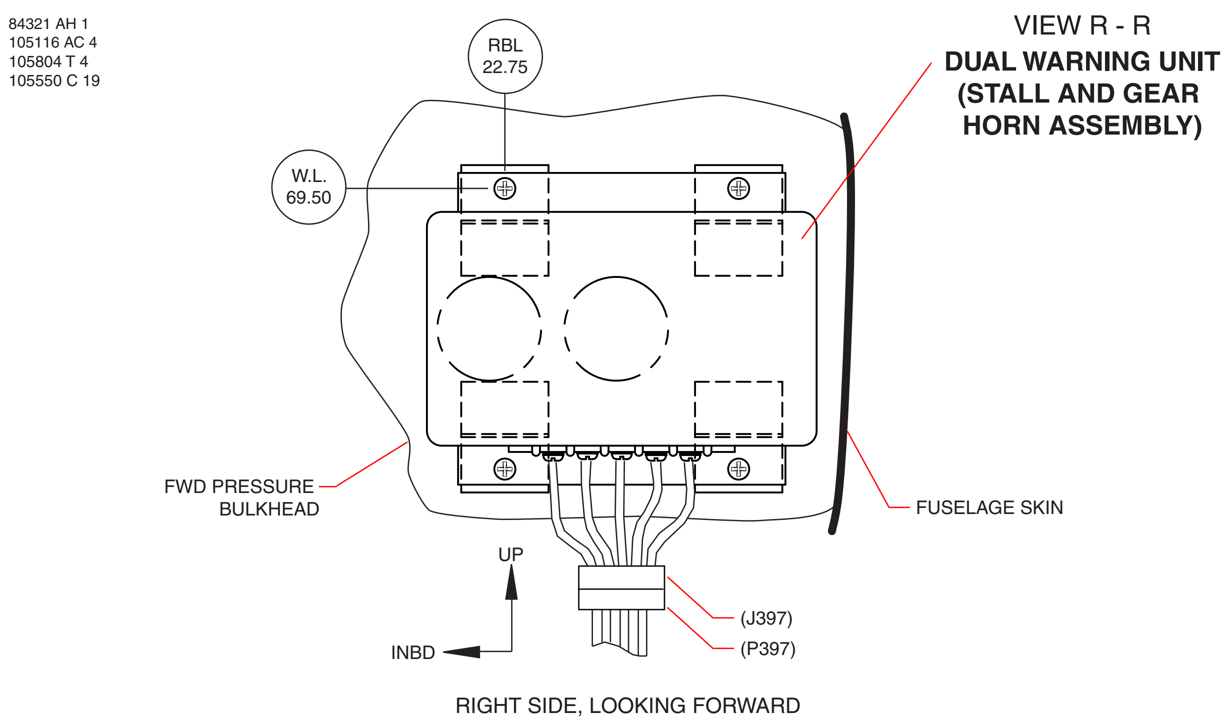
Electrical/Electronic Component Locator
Figure 7 (Sheet 3 of 4)



Electrical/Electronic Component Locator
Figure 7 (Sheet 4 of 4)

Effectivity
with Garmin G1000

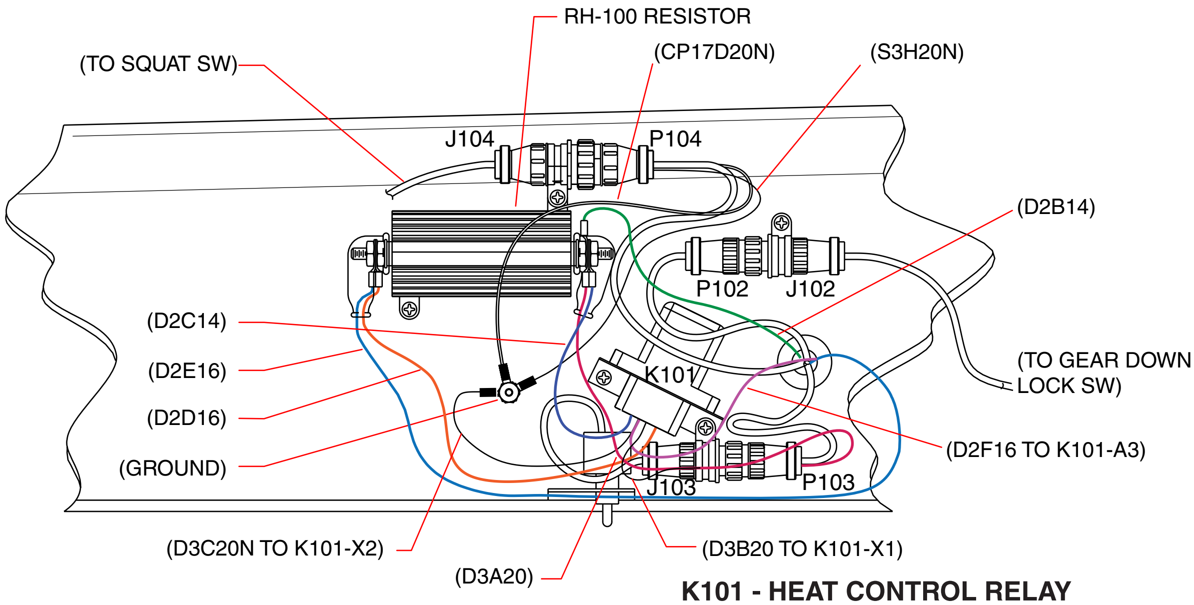




Effectivity
4636375 and up,
4692001 and up

Electrical/Electronic Component Locator
Figure 8

84321 AH 1



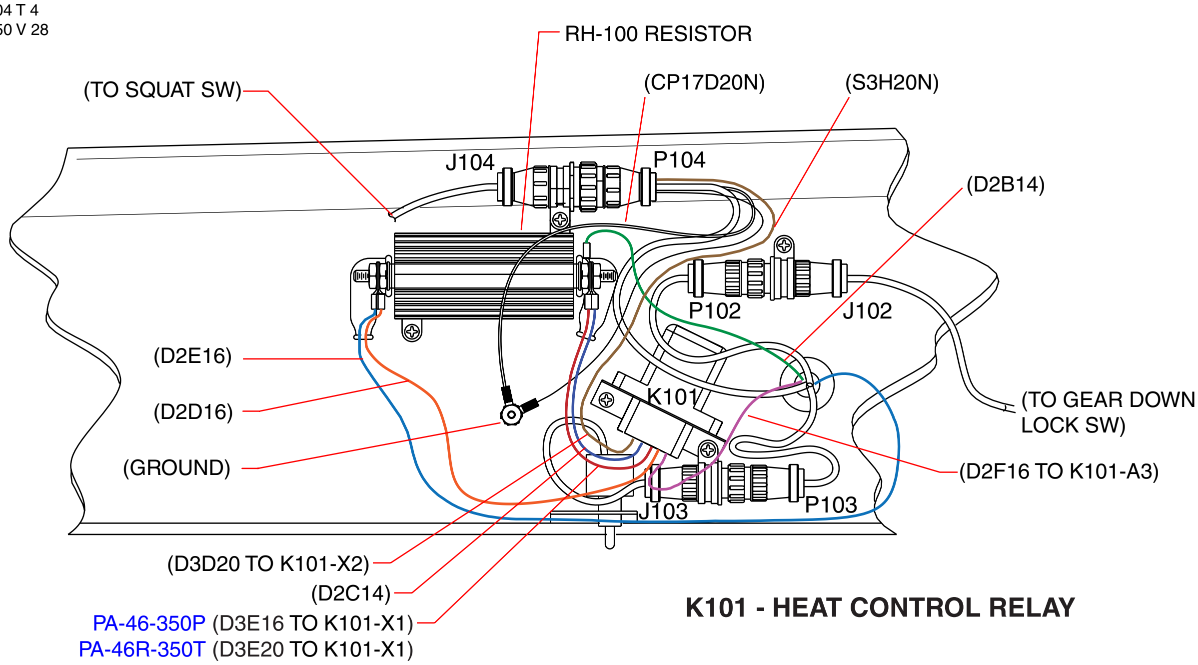
K101 - HEAT CONTROL RELAY

LEFT WHEEL WELL, LOOKING FORWARD AT SPAR

S/N's 4636021 THRU 4636445

S/N's 4636446 AND UP (AND EARLIER PA-46-350P AIRPLANES WITH KIT NO. 88452-002 INSTALLED); AND S/N's 4692001 AND UP

105116 AC 4
105804 T 4
105550 V 28

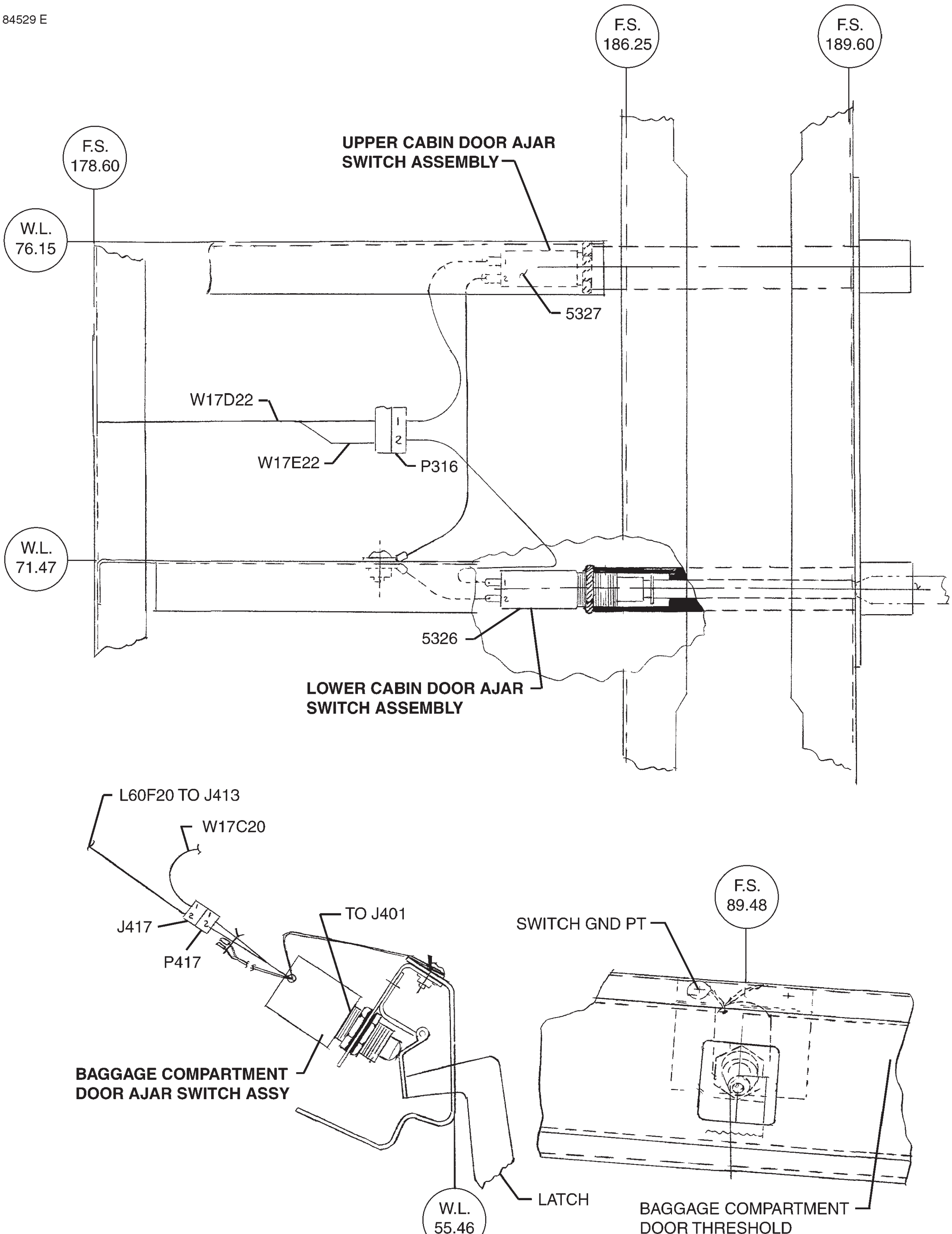


K101 - HEAT CONTROL RELAY

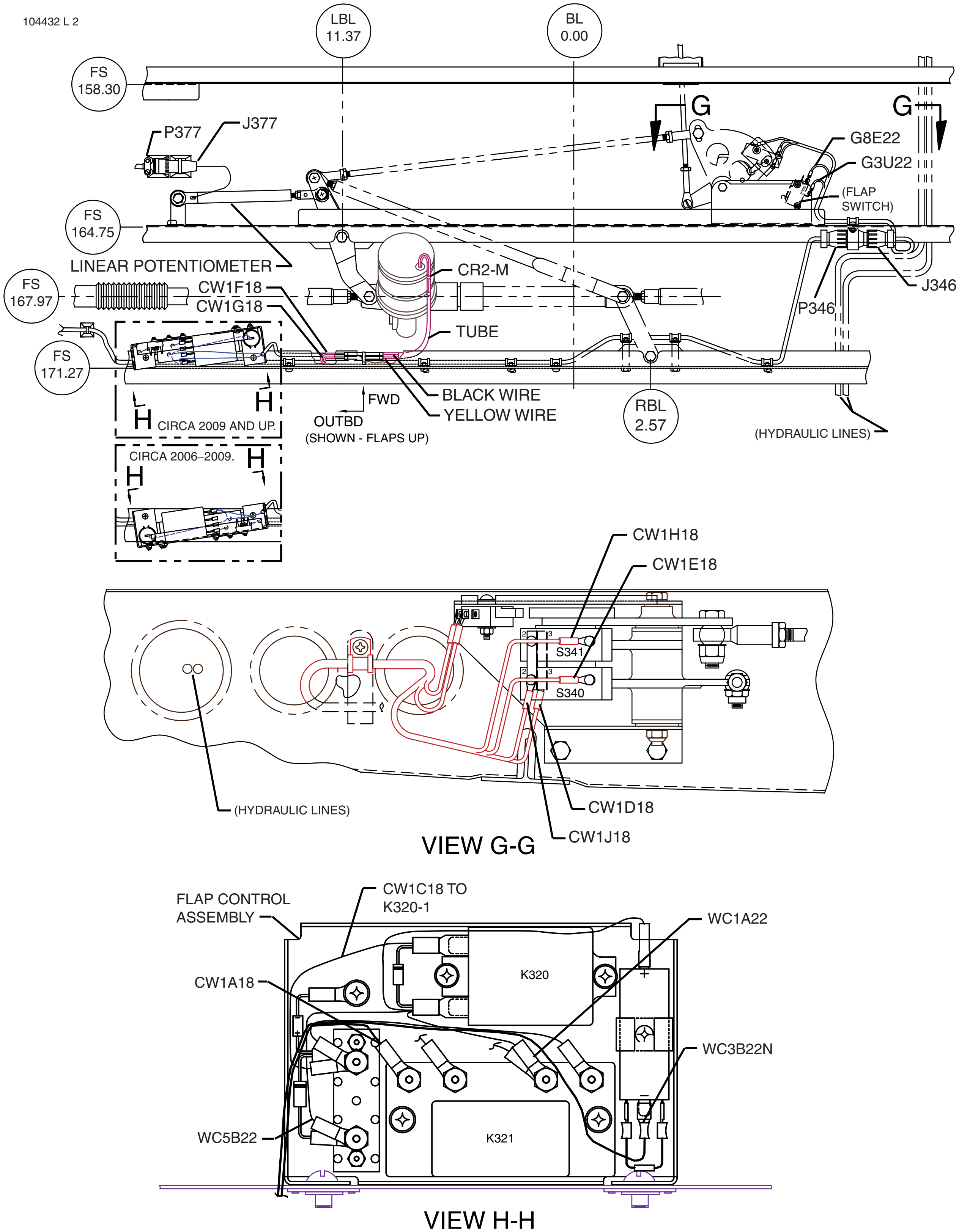
LEFT WHEEL WELL, LOOKING FORWARD AT SPAR

Electrical/Electronic Component Locator
Figure 9

Effectivity
4636021 and up,
4692001 and up

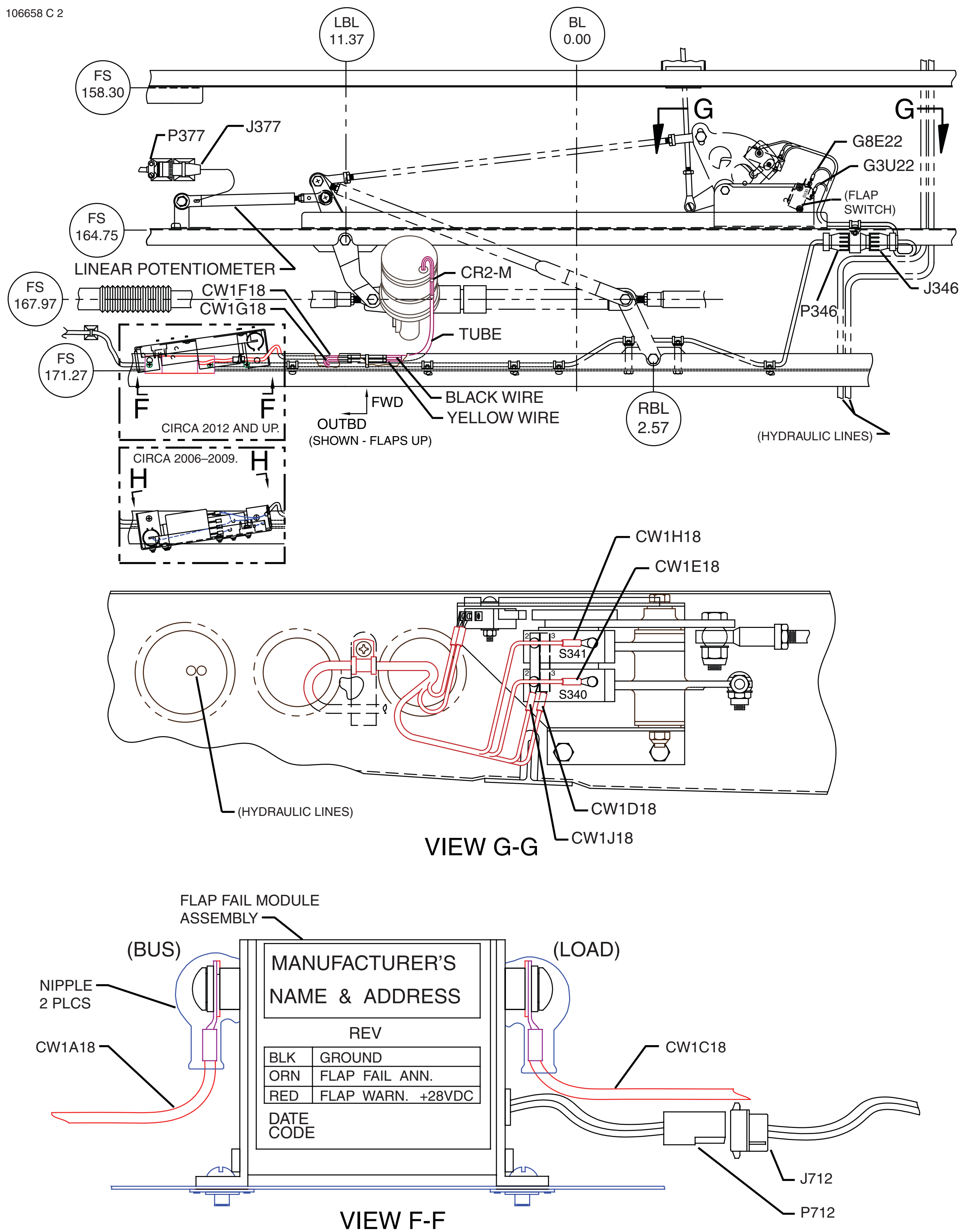


Electrical/Electronic Component Locator
Figure 10



Electrical/Electronic Component Locator
Figure 11 (Sheet 1 of 2)

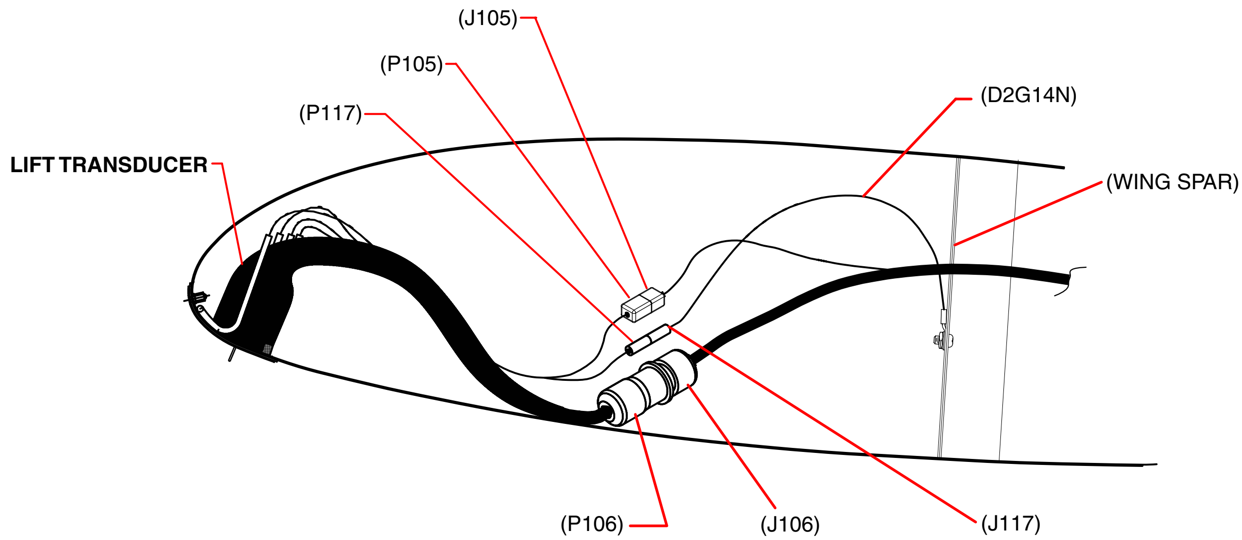
Effectivity
4636375 thru 4636514,
4692001 thru 4692171



Effectivity
4636515 and up,
4692172 and up

Electrical/Electronic Component Locator
Figure 11 (Sheet 2 of 2)

105116 AC 4
105804 T 4

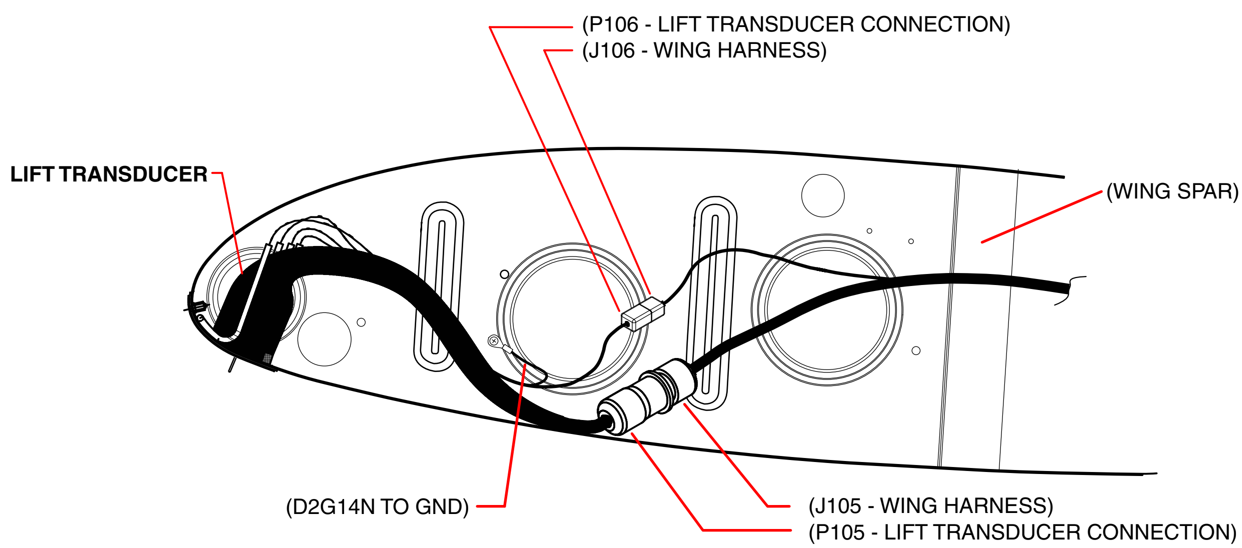


VIEW LOOKING INBOARD

S/N'S 4636001 THRU 4636459, AND 4636461; S/N'S 4692001 THRU 4692133

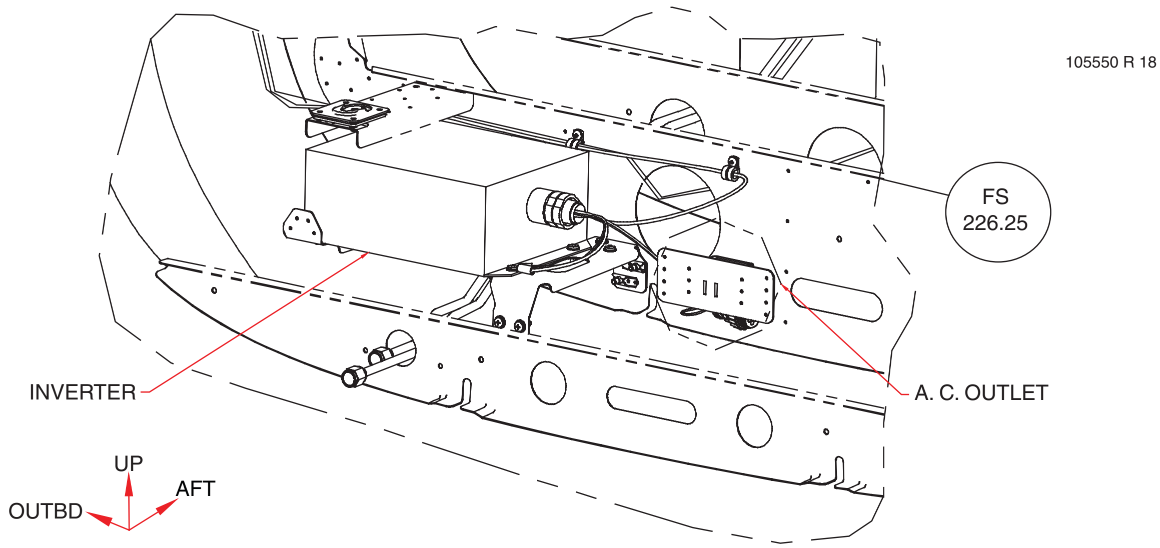
S/N'S 4636460, 4636462 AND UP; S/N'S 4692134 AND UP

105550 V 28



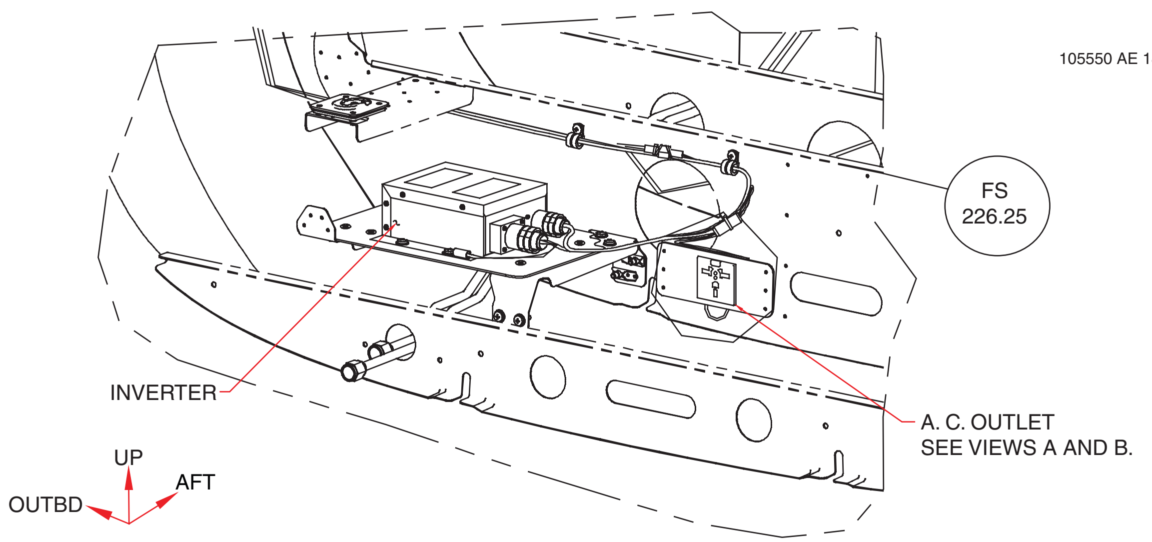
VIEW LOOKING INBOARD

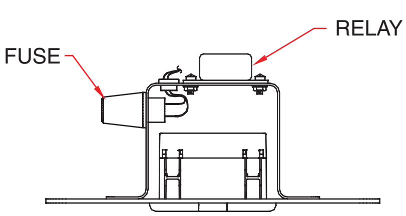
Electrical/Electronic Component Locator
Figure 12



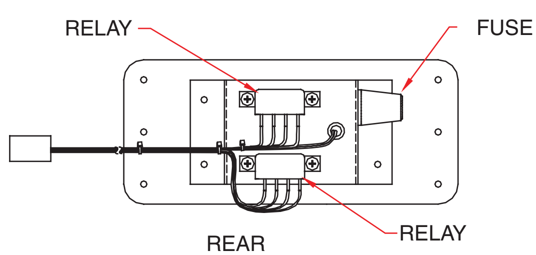
S/N'S 4636460, 4636463 THRU 4636488; S/N'S 4692134 THRU 4692155

S/N'S 4636489 THRU 4636775; S/N'S 4692156 AND UP





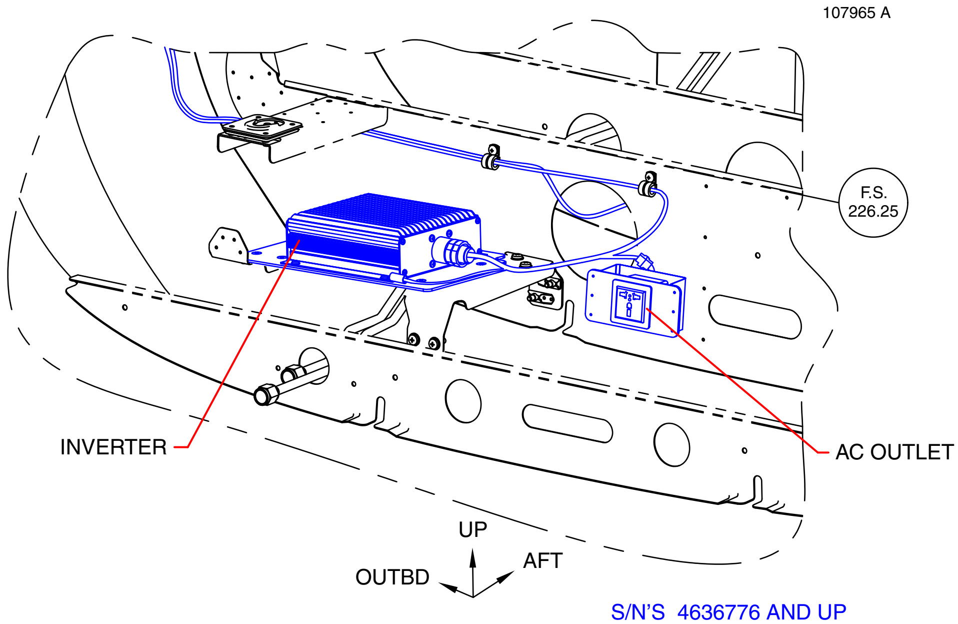
TOP
VIEW A



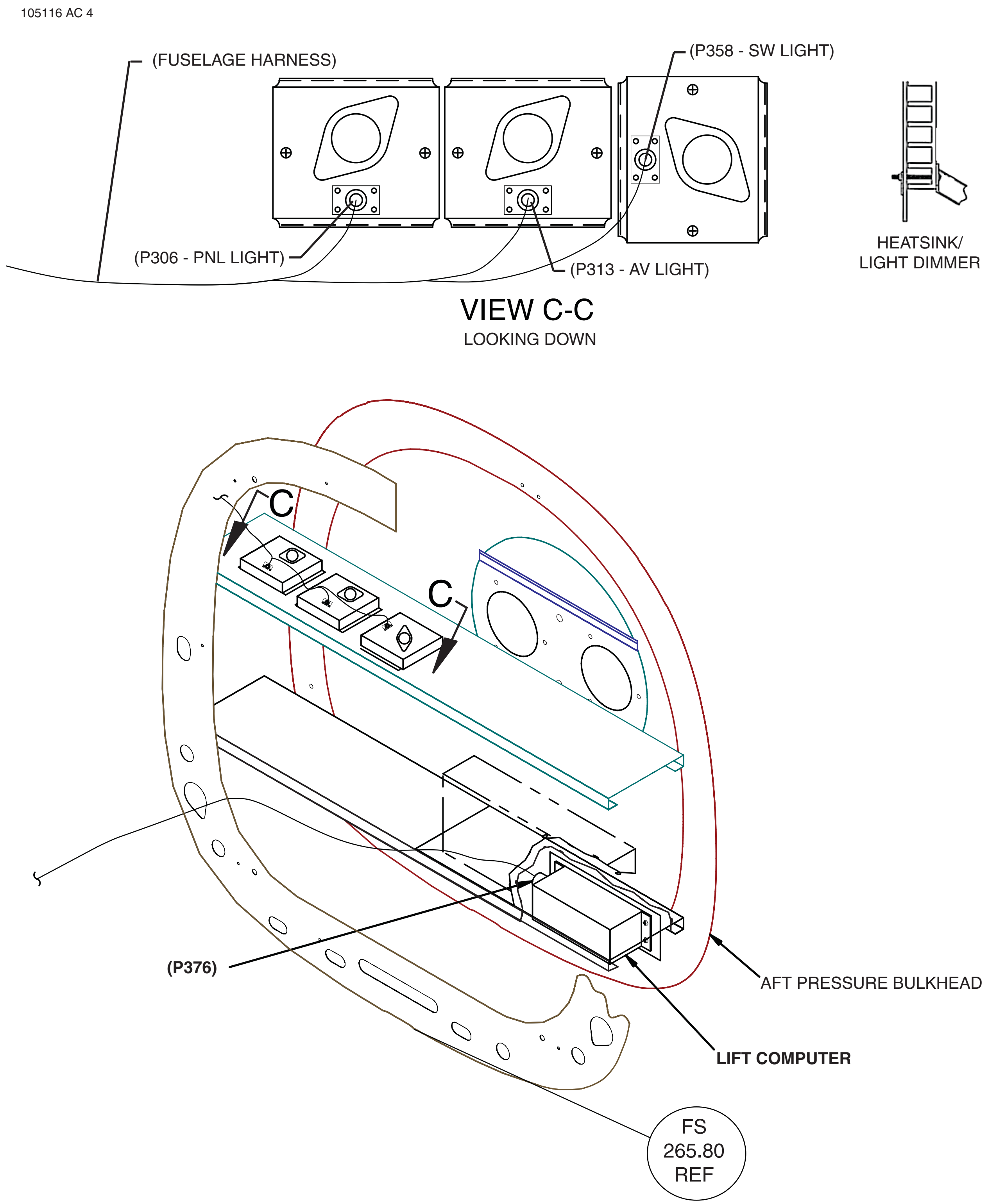
REAR
VIEW B

INVERTER INSTALLATION
LOOKING AFT

Electrical/Electronic Component Locator
Figure 13 (Sheet 1 of 2)

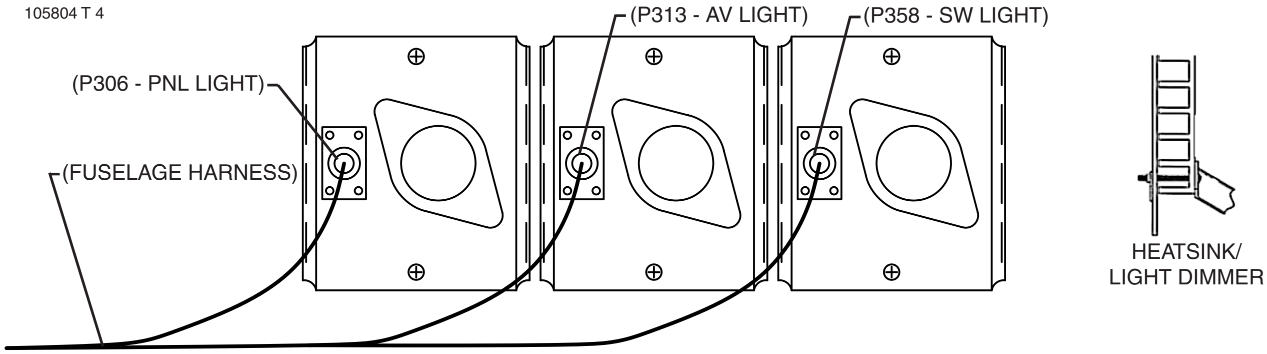


Electrical/Electronic Component Locator
Figure 13 (Sheet 2 of 2)

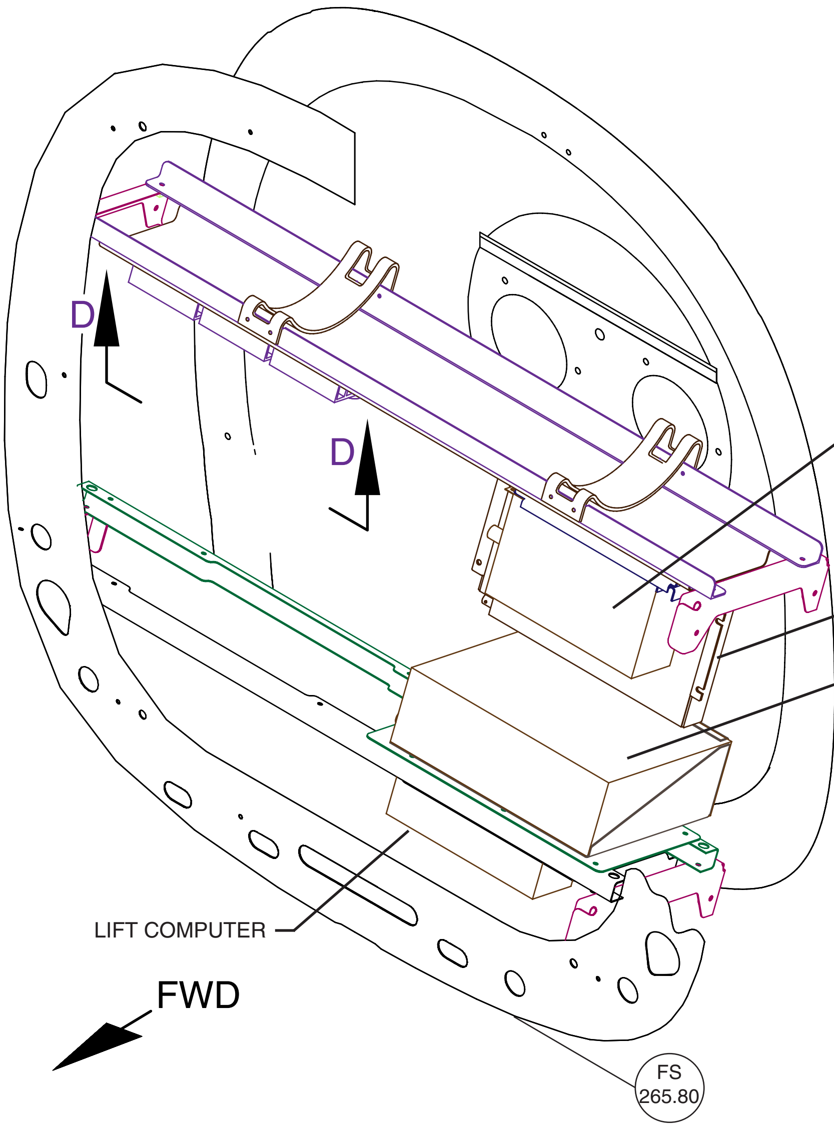


Effectivity
4636001 and up

Electrical/Electronic Component Locator
Figure 14 (Sheet 1 of 2)



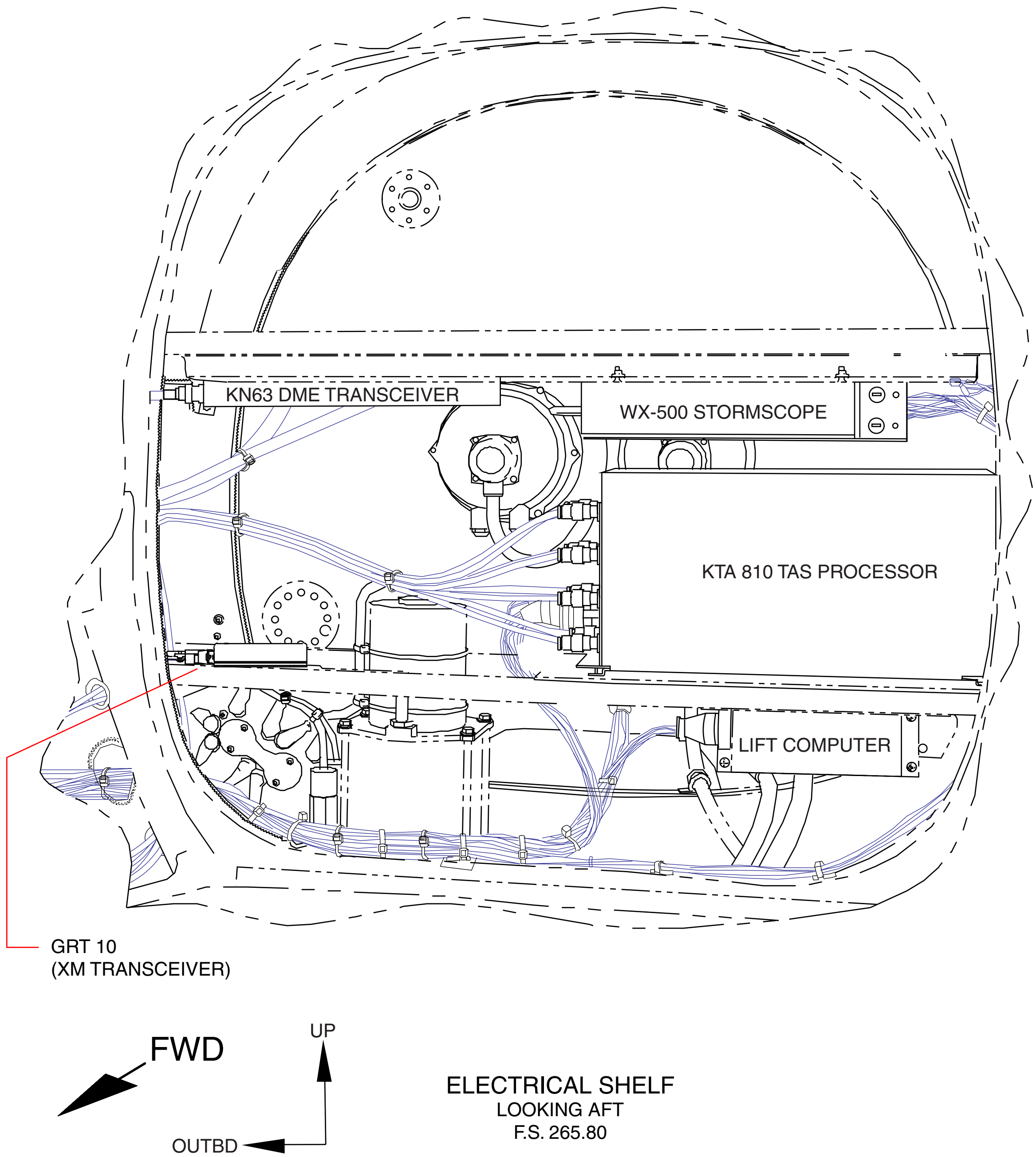
VIEW D-D
LOOKING UP



Electrical/Electronic Component Locator
Figure 14 (Sheet 2 of 2)

Effectivity
4692001 and up

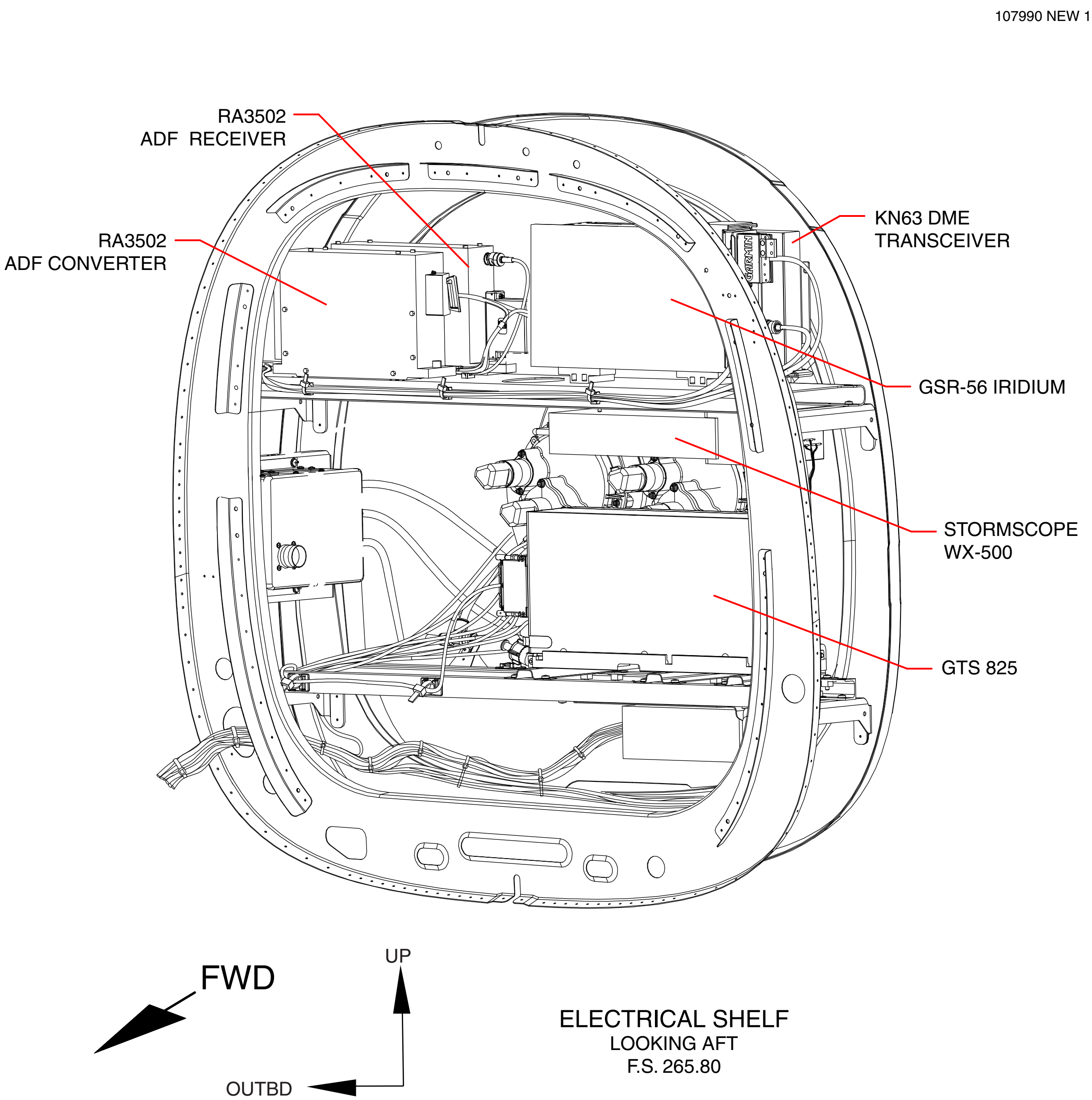
105550 V 14



ELECTRICAL SHELF
LOOKING AFT
F.S. 265.80

Effectivity
with Garmin G1000

Electrical/Electronic Component Locator
Figure 15 (Sheet 1 of 2)



Electrical/Electronic Component Locator
Figure 15 (Sheet 2 of 2)

Effectivity
with Garmin NXi G1000

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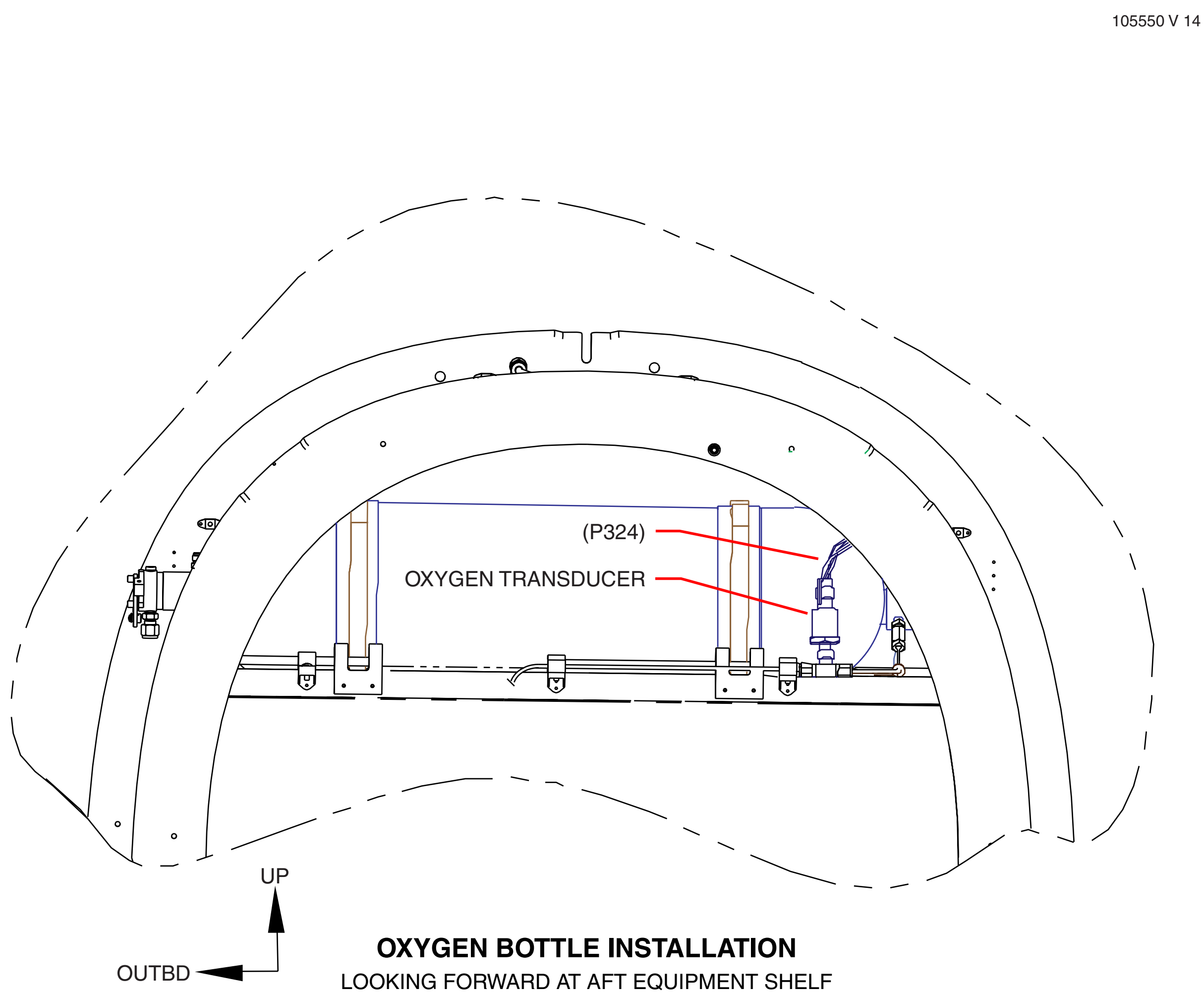
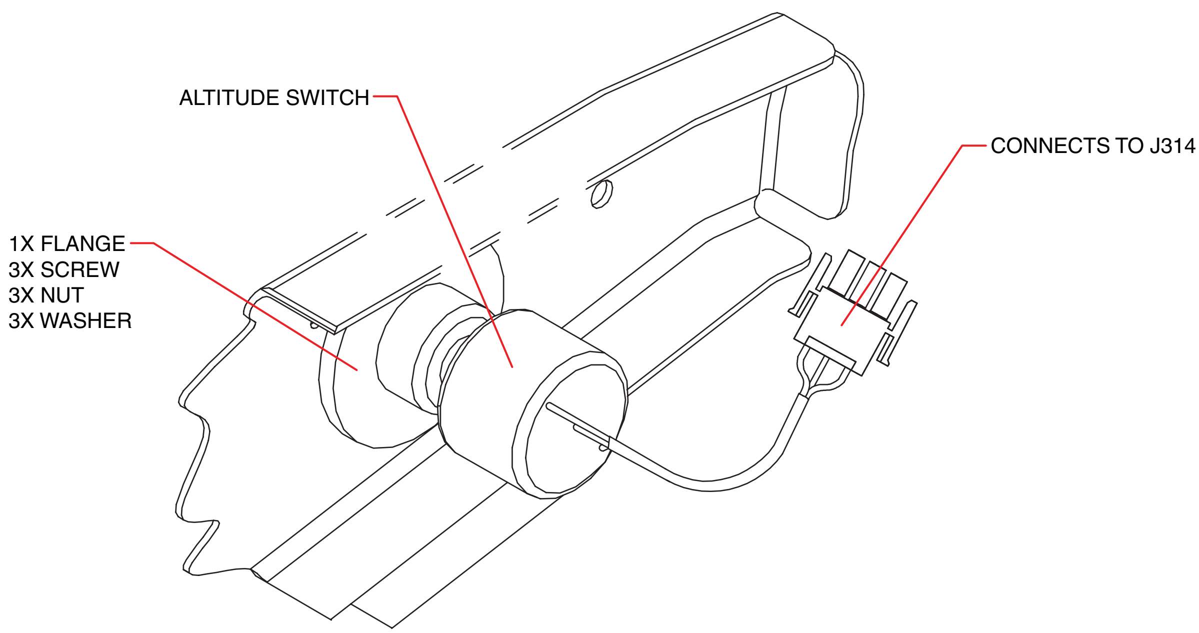


OXYGEN BOTTLE INSTALLATION
LOOKING FORWARD AT AFT EQUIPMENT SHELF

Electrical/Electronic Component Locator
Figure 16

Effectivity
PA-46R-350T only
with Garmin G1000

107950 D



CABIN ALTITUDE SWITCH
LHS LOOKING AFT

Effectivity
4636633, 4636652 and up

Electrical/Electronic Component Locator
Figure 17

CHAPTER

51

STRUCTURES

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CHAPTER 51

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
51-List of	1	Mar 25/21			
Effective Pages	2	Sep 15/09			
51-Table of Contents	1	Mar 25/21			
	2	Nov 30/17			
51-00-00	1	Sep 15/09			
	2	Sep 15/09			
51-10-00	1	Nov 30/17			
	2	Nov 30/17			
	3	Sep 15/09			
	4	Nov 30/17			
	5	Sep 15/09			
	6	Sep 15/09			
51-70-00	1	Dec 15/14			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Sep 15/09			
	5	Sep 15/09			
	6	Sep 15/09			
	7	Sep 15/09			
	8	Sep 15/09			
	9	Sep 15/09			
	10	Jul 1/10			
	11	Sep 15/09			
	12	Sep 15/09			
51-80-00	1	Feb 15/14			
	2	Sep 15/09			
	3	Mar 25/21			
	4	Feb 15/14			
	5	Feb 15/14			
	6	Mar 25/21			
	7	Feb 15/14			
	8	Nov 30/17			
	9	Nov 30/17			
	10	Nov 30/17			
	11	Feb 15/14			
	12	Nov 30/17			
	13	Nov 30/17			
	14	Dec 15/14			

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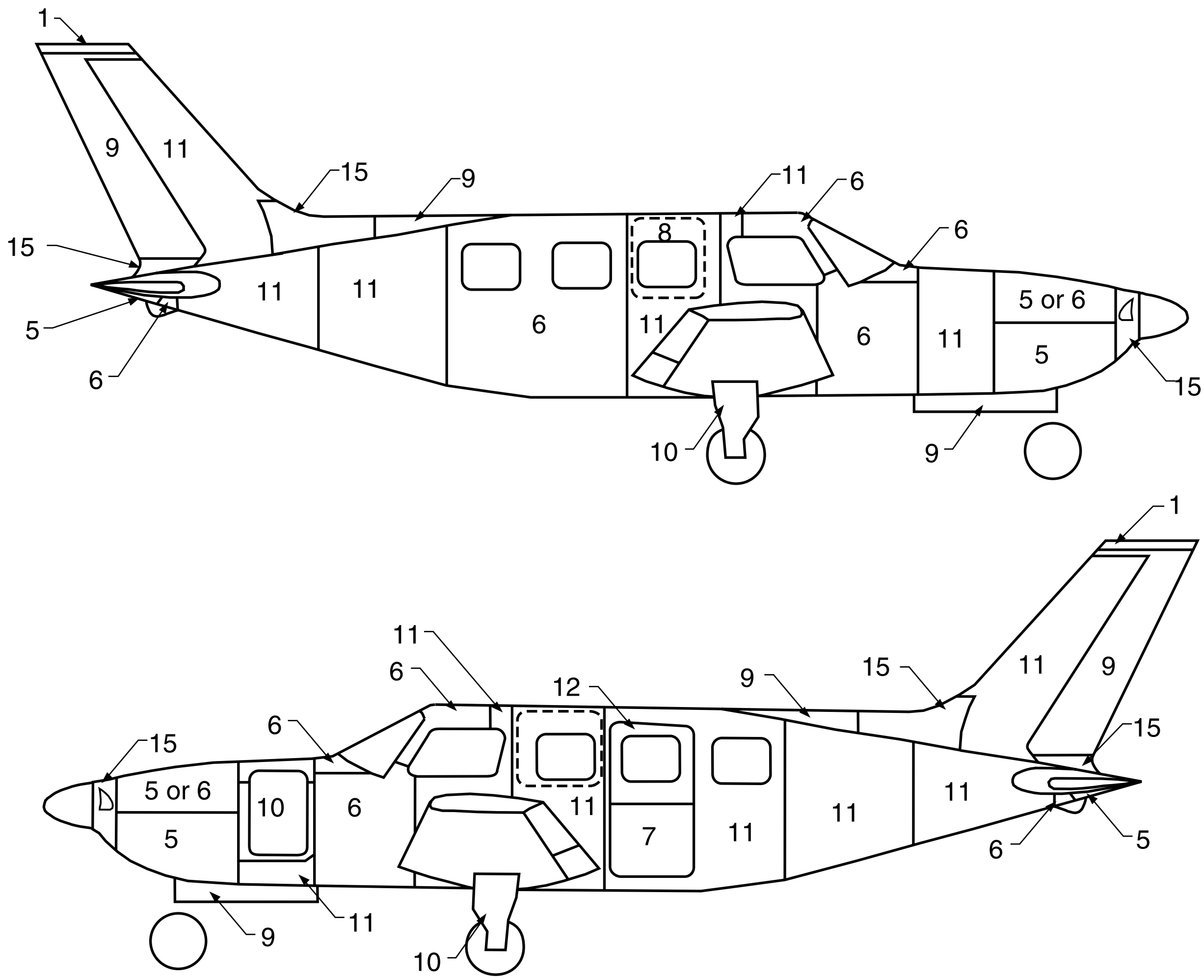
CHAPTER 51 - STRUCTURES

TABLE OF CONTENTS

CHAPTER 51 - STRUCTURES

TABLE OF CONTENTS (continued)

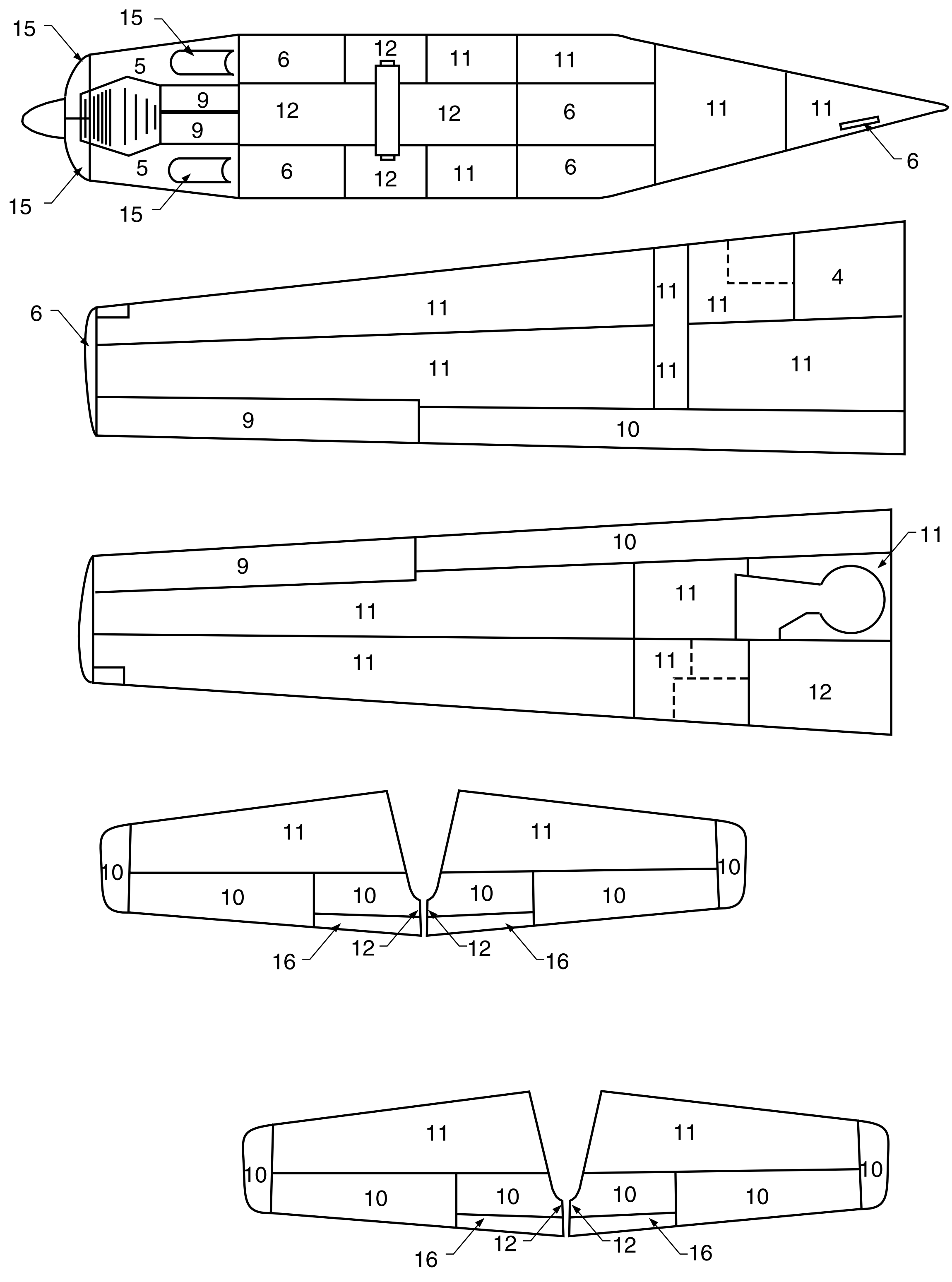
GENERAL



NUMBER	MATERIAL	THICKNESS
1	2024-0	.020
2	2024-0	.025
3	2024-0	.032
4	2024-0	.040
5	2024-0*	.025
6	2024-0*	.032
7	2024-0*	.040
8	2024-0*	.063
9	2024-T3	.020
10	2024-T3	.025
11	2024-T3	.032
12	2024-T3	.040
13	2024-T3*	.040
14	5052-H34	.025
15	FIBERGLASS	
16	2024-T3	.016

* HEAT TREAT TO T42

Skin Material Specifications
Figure 17 (Sheet 1 of 2)



Skin Material Specifications
Figure 1 (Sheet 2 of 2)

INVESTIGATION, CLEANUP AND AERODYNAMIC SMOOTHNESS

1. Corrosion Control

Corrosion is the deterioration of metal by chemical or electrochemical attack. Water which is allowed to remain on the aircraft and industrial pollution are the major causes of corrosion in aircraft. The two general types of corrosion are:

Direct chemical attack (i.e. spilled battery acid).

Electrochemical attack which requires a medium (usually water).

The latter is the most common and is responsible for most forms of aircraft corrosion.

Since corrosion is a constant threat, the only effective method to control it is a routine of regular inspection, cleaning, and surface refinishing.

A. Forms of Corrosion

See "Chart 1" on page 51102.

The following are the most common forms of corrosion:

(1) Surface Corrosion appears as a general roughening or pitting on the surface usually accompanied by a powdery deposit of corrosion products. It may spread under the surface and not be recognized until the paint or plating is lifted off the surface in small blisters.

(2) Dissimilar Metal Corrosion may occur when two dissimilar metals are contacting each other. This type may be serious because it usually takes place out of sight. The only way to find it before structural failure is by disassembly and inspection. Insulating is necessary between two contacting dissimilar surfaces (2-3 coats of waterborne, chromated, fluid resistant, epoxy primer (i.e., PRC Desoto) or a coat of epoxy polyamide on each surface; plus a .003 thick piece of vinyl tape if one of the surfaces is magnesium).

(3) Intergranular Corrosion is difficult to detect in its early stages. When severe, it causes the surface of the metal to exfoliate (flake or lift).

(4) Stress Corrosion is the result of sustained tensile stresses and corrosive environment. It usually occurs in assemblies such as aluminum alloy bellcranks with pressed-in bushings; landing gear shock struts with pipe thread grease fittings, clevis pin joints and shrink-fit parts.

(5) Fretting Corrosion takes place when two parts rub together, constantly exposing fresh active metal to the corrosive effects of the atmosphere.

(6) Filiform Corrosion is the appearance of numerous meandering thread-like filaments of corrosion on the surface of various types of metal.

B. Conditions Affecting Corrosion

Some conditions which affect the occurrence of corrosion are:

(1) Heat and humidity increase corrosion.

(2) Different (i.e. - dissimilar) metals and their relative sizes affect resistance or susceptibility to corrosion.

(3) Frequent contributing factors to corrosion:

(a) Soil and atmosphere dust.

(b) Oil, grease, and exhaust residues.

(c) Salt water and salt moisture condensation.

(d) Spilled battery acids and caustic cleaning solution.

(e) Welding, brazing, and soldering flux residue.

CHART 1
TYPES OF METAL CORROSION

Type of Material	Type of Corrosion	Remedy [2]
Steel.	Rust [1].	Complete removal of corrosion by mechanical means.
Aluminum.	White to grey powdery material.	Mechanical polishing or brushing with material softer than aluminum.
Magnesium (highly susceptible to corrosion).	White powdery snow-like mounds and white spots.	Mechanical polishing or brushing for a smooth finish.
Cadmium (plating).	White to brown to black mottling of surface (plating is still protecting until iron appears).	Mechanical removal of corrosion is limited to metal surfaces from which cadmium has been depleted.
Chromium (plating).	May pit in chloride environment.	Polishing and buffing.

(1) Red rust generally shows on bolts, nuts, and other aircraft hardware. Rust in these areas is generally not dangerous, however, it shows a need for maintenance and the possibility of corrosive attack in more critical areas. Any surface corrosion on highly stressed steel parts is potentially dangerous. A careful removal of corrosion using mild abrasives (rouge or fine grit aluminum oxide paper) is necessary. Do not overheat metal when removing corrosion.

(2) For abrasion, do not use dissimilar material (for example steel wool on aluminum). Remove only material required to clean affected area.

(4) A clean aircraft will resist corrosion better than a dirty one. Cleaning frequency depends on several factors, including geographical location, type of operation, etc. Remove soil as soon as possible, especially when in a high-temperature area.

(5) After cleaning, verify that no cleaning solution remains in any holes, crevices, or joints as it may lead to increased corrosion. All exposed areas (landing gear, flap tracks, control surface, hinge parts, etc) must be lubricated after cleaning.

C. Corrosion Inspection

CAUTION: INSPECTION FOR CORROSION MUST BE PERFORMED BY PERSONS FAMILIAR WITH CORROSIVE PROBLEMS AND REMEDIES.

NOTE: Some areas of the airplane have been treated with a corrosion inhibiting compound which requires re-treatment at five (5) year intervals. See Each Five (5) Years, Per Calendar Year, 5-30-00.

Check for corrosion at every inspection. In trouble areas, inspection frequency must be increased. In addition to routine inspections:

(1) Aircraft operating around a marine environment must be given special inspections on a weekly basis. See Per Specific Operation / Operating Environment, 5-30-00.

(2) Aircraft operating in semi-acid conditions must be inspected monthly. Semi-acid conditions are likely to occur in industrialized areas where sulphur-bearing particles in dust, smoke, and smog will attack painted surfaces. See Per Specific Operation / Operating Environment, 5-30-00.

(3) Inspection for corrosion must be performed by personnel familiar with corrosive problems and remedies.

(a) Daily and preflight inspection must include engine frontal areas, all intake vents, engine compartments, gaps, seams, and faying surfaces in exterior skins, wheel and wheel well areas, battery compartment, fuel cell, all other drains, and any bilge areas not requiring extensive removal of inspection access covers.

(b) Detailed inspection must include above referenced areas along with areas requiring removal of inspection plates and panels to thoroughly inspect internal cavities of aircraft.

(4) Paint tends to hide corrosion in its initial stages. The results of corrosion can sometimes be seen as blisters, flakes, chips, and other irregularities in paint.

D. Corrosion Removal and Control

CAUTION: THE DEPTH OF MATERIAL REMOVED MUST NOT EXCEED SAFE LIMITS.

CAUTION: REMOVAL OF SEVERE CORROSION MAY BE CONSIDERED A MAJOR REPAIR. ANY REPAIR OF THIS TYPE MUST BE APPROVED BY THE FAA BEFORE THE AIRPLANE CAN BE RETURNED TO SERVICE.

Corrosion cannot be prevented or eliminated on aircraft; it can only be reduced to an acceptable level by proper control methods.

All corrosion products must be removed prior to refinishing. If not removed, corrosion will begin again, even though affected area is refinished.

(1) Before beginning any rework:

(a) Position airplane in a wash rack or provide some type of washing apparatus for rapid rinsing of all surfaces.

(b) Connect static ground line to airplane.

(c) Remove airplane battery if required.

(d) Protect pitot-static ports, engine openings, airscoops, louvers, wheels, tires, and other portions of airplane from moisture and chemical brightening agents.

(e) Protect surfaces next to rework areas from chemical paint strippers, corrosion removal agents, and surface treatment materials.

(2) Evaluate corrosion damage to determine type and extent of repairs required. Proceed as follows:

(a) Light Corrosion: discoloration or pitting. Remove by light hand sanding or a small amount of chemical treatment.

(b) Moderate Corrosion: similar to light corrosion except there is blistering or evidence of scaling and flaking. Remove by extensive hand or mechanical sanding.

(c) Severe Corrosion: similar to moderate corrosion with severe blistering, exfoliation, scaling, or flaking. Remove by extensive mechanical sanding or grinding.

E. Corrosion Prone Areas

Certain areas are more prone to corrosion than others. The following list is a general guide to areas where corrosion is frequently found.

(1) Areas around steel fasteners are susceptible to corrosion. The paint on these areas cracks which allows moisture to seep in and corrode the underlying metal. Each time the fastener is removed, it should be coated with Fluid Resistant Epoxy Primer (or equivalent) before reinstallation. The paint should be wet when the fastener is installed.

(2) Fluids tend to seep into faying surfaces, seams and joints due to capillary action. The effect of this type of intrusion is usually detectable by irregularities in the skin's surface.

(3) Spot welded assemblies are particularly prone to corrosion. The only means to prevent this type of corrosion is by keeping potential moisture entry points in the spot weld filled with a sealant or preservative compound. On an aluminum spot welded assembly, a chromate conversion coating before paint is applied will help prevent corrosion.

(4) Areas exposed to exhaust gases may have their finish damaged by deposits. These deposits may result in an aggressive attack on the metal by corrosion. Heat from the exhaust may also blister or otherwise damage the paint. Gaps, seams, hinges and fairings are some places where exhaust gas deposits may be trapped and not reached by normal cleaning methods.

(5) The wheel well and landing gear are the most exposed parts of the aircraft. Due to the complexity of its shape, assemblies and fittings, maintaining a protective coverage is difficult. The especially troublesome areas are:

(a) Magnesium wheels: around bolt heads, lugs and wheel well areas:

(b) Exposed rigid tubing, B-nuts, ferrules, under clamps and tubing identification tape:

(c) Exposed position indicator switches and other electrical equipment:

(d) Crevices between stiffeners, ribs and lower skin surfaces.

(6) Flaps, flight control slots and equipment installed in these areas may corrode unnoticed unless a careful surveillance is maintained.

(7) Engine frontal areas, air inlet ducts and the leading edge of wings, because they are constantly exposed to abrasion by dirt, dust, gravel and rain, should be checked frequently for the beginning of corrosion.

(8) Hinges (piano hinges especially) are extremely vulnerable to corrosion. Their protective coatings wear away and they naturally trap dirt, salt and moisture.

(9) Control cables may have bare spots in their preservative coating which will lead to corrosion. Cables having external corrosion must be checked for internal corrosion. If internal corrosion is present, replace the cable. If only external corrosion is present, remove corrosion with wire brush and recoat cable with preservative.

(10) Check and clean drain holes regularly.

(11) Battery compartment and vent openings are especially prone to corrosion due to spilled electrolyte. Fumes from overheated battery electrolyte will spread to adjacent areas and cause rapid corrosion of unprotected surfaces. Frequent cleaning and neutralization of deposits will minimize corrosion in this area.

(12) Magnesium parts are prone to corrosion. Special attention must be given to their surface treatment, proper insulation (due to dissimilar metal corrosion), and paint coatings.

(13) Electrical components and connectors must be checked. Inspection frequency is based on operational environment and past trouble.

(14) Skin joints and lap-overs are two areas that can trap and hold moisture. Corrosion in these areas may go unnoticed unless particular attention is paid to them during inspection.

(15) Hoses, having an internal wire braid, which are located in a position where they are frequently water soaked, need a protective treatment.

(16) Drilled holes and trimmed ends of sandwich panels must be protected. Use an inhibitor solution or sealant application. Any gaps or cavities which allow dirt or moisture to enter must be filled with sealant.

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REPAIRS

WARNING: NO ACCESS HOLES ARE PERMITTED IN ANY CONTROL SURFACE.

WARNING: USE OF PATCH PLATES FOR REPAIRS OF ALL MOVABLE CONTROL SURFACES IS PROHIBITED. USE OF ANY FILLER MATERIAL NORMALLY USED FOR REPAIR OF MINOR DENTS AND/OR MATERIALS USED FOR FILLING INSIDE OF SURFACES IS ALSO PROHIBITED ON ALL MOVABLE CONTROL SURFACES.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

Structural repair methods used for minor repairs must be in accordance with FAA Advisory Circular 43.13-1, latest revision. To assist in making repairs, Figure 1, 51-00-00, identifies type and thickness of various skin material used.

Repairs to areas defined in FAR Part 43, Appendix A, must be shown (using approved engineering data) to not diminish strength or the Fatigue Life of the component, if a life limit is stated herein (see 4-00-00) or in the aircraft type certificate data sheet (TCDS).

When a repair is proposed, it is the responsibility of the repairer per AC 43.13-1 to determine that the proposed repair is not contrary to manufacturer's data. The repairer or aircraft owner or his agent should contact Piper directly to determine that a proposed repair is not in conflict with minimum type design capability.

Temporary repairs, when required, must add Instructions for Continued Airworthiness (ICA) to the maintenance record. Any such ICA must be based on approved data.

1. Fiberglass Repairs

The repair procedure in this section will describe the methods for the repair of fiberglass reinforced structures. This section describes Touch-up and Surface Repairs such as blisters, open seams, delaminations, cavities, small holes and minor damages that have not harmed the fiberglass cloth material. Also covered are Fracture and Patch Repairs such as puncture, breaks and holes that have penetrated through the structure and damaged the fiberglass cloth. A repair kit, Piper P/N 766-222, that contains the necessary material for such repairs, is available through Piper Distributors.

NOTE: Very carefully follow resin and catalyst mixing instructions furnished with repair kit.

A. Touch-up and Surface Repairs

(1) Remove wax, oil, and dirt from around the damaged area, using acetone, Methylethylketone, or equivalent, and remove paint to gel coat.

(2) The damaged area may be scraped with a fine blade knife or a power drill with a burr attachment to roughen the bottom and sides of the damaged area. Feather the edge surrounding the scratch or cavity. Do not undercut the edge. (If the scratch or cavity is shallow and penetrates only the surface coat, continue to step (8)

(3) Pour a small amount of resin into a jar lid or on a piece of cardboard, just enough to fill the area being worked on. Mix an equal amount of milled fiberglass with the resin, using a putty knife or stick. Add catalyst, according to kit instruction, to the resin and mix thoroughly. A hypodermic needle may be used to inject gel into small cavities not requiring fiberglass millings mixed with the gel.

(4) Work the mixture of resin, fibers and catalyst into the damaged area, using the sharp point of a putty knife or stick to press it into the bottom of the hole and to puncture any air bubbles which may be present. Fill the scratch or hole above the surrounding undamaged area about .062 of an inch.

(5) Lay a piece of cellophane or waxed paper over the repair to cut off air and start the cure of gel mixture.

(6) Allow the gel to cure 10 to 15 minutes until it feels rubbery to the touch. Remove the cellophane and trim flush with the surface, using a sharp razor blade or knife. Replace the cellophane and allow to cure completely for 30 minutes to an hour. The patch will shrink slightly below the structure surface as it cures. (If wax paper is used, make sure the wax is removed from surface.)

(7) Rough up the bottom and edges of the hole with the electric burr attachment or rough sandpaper. Feather hole into surrounding gel coat, do not undercut.

(8) Pour out a small amount of resin, add catalyst and mix thoroughly, using a cutting motion rather than stirring. Use no fibers.

(9) Using the tip of a putty knife or fingertips, fill the hole to about .062 of an inch above the surrounding surface with the gel coat mixture.

(10) Lay a piece of cellophane over the patch to start the curing process. Repeat step (6), trimming patch when partially cured.

(11) After trimming the patch, immediately place another small amount of gel coat on one edge of the patch and cover with cellophane. Then, using a squeegee or the back of a razor blade, level with area surrounding the patch; leave the cellophane on patch for one to two hours or overnight, for complete cure.

(12) After repair has cured for 24 hours, sand patched area, using a sanding block with fine wet sandpaper. Finish by priming, again sanding and applying color coat.

B. Fracture and Patch Repairs

(1) Remove wax, oil and dirt from around the damaged area with acetone, methylethylketone or equivalent.

(2) Using a key hole saw, electric saber saw, or sharp knife cut away ragged edges. Cut back to sound material.

(3) Remove paint three inches back from around damaged area.

(4) Working inside the structure, bevel the edges to approximately a 30 degree angle and rough-sand the hole and the area around it, using 80 grit dry paper. Feather back for about two inches all around the hole. This roughens the surface for strong bond with patch.

(5) Cover a piece of cardboard or metal with cellophane. Tape it to the outside of the structure, covering the hole completely. The cellophane should face toward the inside of the structure. If the repair is on a sharp contour or shaped area, a sheet of aluminum formed to a similar contour may be placed over the area. The aluminum should also be covered with cellophane.

(6) Prepare a patch of fiberglass mat and cloth to cover an area two inches larger than the hole.

(7) Mix a small amount of resin and catalyst; enough to be used for one step at a time, according to kit instructions.

(8) Thoroughly wet mat and cloth with catalyzed resin. Daub resin on mat first, and then on cloth. Mat should be applied against structure's surface with cloth on top. Both pieces may be wet out on cellophane and applied as a sandwich. Enough fiberglass cloth and mat reinforcements should be used to at least replace the amount of reinforcements removed in order to maintain the original strength. If damage occurred as a stress crack, an extra layer or two of cloth may be used to strengthen area.

(9) Lay patch over hole on inside of structure, cover with cellophane, and squeegee from center to edges to remove all air bubbles and assure adhesion around edge of hole. Air bubbles will show white in the patch and they should all be worked out to the edge. Remove excess resin before it gels on the part. Allow patch to cure completely.

(10) Remove cardboard or aluminum sheet from outside of hole and rough-sand the patch and edge of hole. Feather edge of hole about two inches into undamaged area.

(11) Mask area around hole with tape and paper to protect surface. Cut a piece of fiberglass mat about one inch larger than the hole and one or more pieces of fiberglass cloth two inches larger than the hole. Brush catalyzed resin over hole, lay mat over hole and wet out with catalyzed resin. Use a daubing action with brush. Then apply additional layer or layers of fiberglass cloth to build up patch to the surface of structure. Wet out each layer thoroughly with resin.

(12) With a squeegee or broad knife, work out all air bubbles in the patch. Work from center to edge pressing patch firmly against the structure. Allow patch to cure for 15 to 20 minutes.

(13) As soon as the patch begins to set, but while still rubbery, take a sharp knife and cut away extra cloth and mat. Cut on outside edge of feathering. Strip cut edges of structure. Do this before cure is complete, to save extra sanding. Allow patch to cure overnight.

(14) Using dry 80 grit sandpaper on a power sander or sanding block, smooth patch and blend with surrounding surface. Should air pockets appear while sanding, puncture and fill with catalyzed resin. A hypodermic needle may be used to fill cavities. Let cure and resand.

(15) Mix catalyzed resin and work into patch with fingers. Smooth carefully and work into any crevices.

(16) Cover with cellophane and squeegee smooth. Allow to cure completely before removing cellophane. Let cure and resand.

(17) Brush or spray a coat of catalyzed resin to seal patch. Sand patch, finish by priming, again sanding and applying color coat.

NOTE: Brush and hands may be cleaned in solvents such as acetone or methylethylketone. If solvents are not available, a strong solution of detergent and water may be used.

2. Thermoplastic Repairs

The following procedure will assist in making field repairs to items made of thermoplastic which are used throughout the airplane. “Chart 1” lists materials needed to perform these repairs along with suggested suppliers. Common safety precautions should be observed when handling some of the materials and tools used while making these repairs. Refer to Vendor Information, Chapter 91, for supplier addresses.

A. Surface Preparation

(1) Surface dirt and paint if applied must be removed from the item being repaired. Household cleaners have proven most effective in removing surface dirt.

(2) Preliminary cleaning of the damaged area with perchlorethylene or V M & P Naphtha will generally ensure a good bond between epoxy compounds and thermoplastic.

B. Surface Scratches, Abrasion or Ground-in-Dirt

See “Figure 1”.

(1) Shallow scratches and abraded surfaces are usually repaired by following directions on containers of conventional automotive buffing and rubbing compounds.

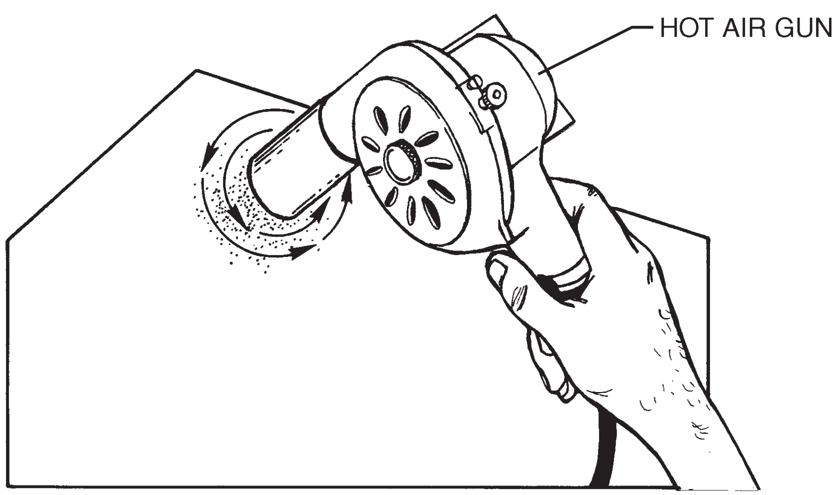
(2) If large dirt particles are embedded in thermoplastic parts they can be removed with a hot air gun capable of supplying heat in the temperature range of 300° to 400° F. Use care not to overheat the material. Hold the nozzle of the gun about 1/4 of an inch away from the surface and apply heat with a circular motion until the area is sufficiently soft to remove the dirt particles.

(3) The thermoplastic will return to its original shape upon cooling.

CHART 1
LIST OF MATERIALS (THERMOPLASTIC REPAIRS)

Items	Descriptions	Suppliers
Buffing and Rubbing Compounds	Automotive Type - Axalta (formerlyDuPont) #7	Axalta Company
	Ram Chemical #69 x 1	Ram Chemicals
	Mirror Glaze #1	Mirror Bright Polish Co., Inc.
Cleaners	Fantastic Spray	Obtain From Local Suppliers
	Perchlorethylene	
	V M & P Naphtha (Lighter Fluid)	
ABS-Solvent Cements	Solarite #11 Series	Solar Compounds Corp.
Solvents	Methylethylketone	Obtain From Local Suppliers
	Methylene Chloride	
	Acetone	
Epoxy Patching Compound	Solarite #400	Solar Compounds Corp.
Hot Melt Adhesives	Stick Form 1/2 in. dia.	Sears Roebuck & Co. or
Polyamids and Hot Melt Gun	3 in. long	Most Hardware Stores
Hot Air Gun	Temp. Range 300° to 400° F	Local Suppliers

871



Surface Scratches - Abrasions or Ground-in Dirt
Figure 1

C. Deep Scratches, Shallow Nicks and Small Holes - Less than 1 inch in diameter

See “Figure 2”.

(1) Solvent cements will fit virtually any of these applications. If the area to be repaired is very small, it may be quicker to make a satisfactory cement by dissolving thermoplastic material of the same type being repaired in solvent until the desired paste like consistency is achieved.

(2) This mixture is then applied to the damaged area. Upon solvent evaporation, the hard durable solids remaining can easily be shaped to the desired contour by filing or sanding.

(3) Solvent adhesives are not recommended for highly stressed areas, or thin walled parts or for patching holes greater than 1/4 inch in diameter.

(4) For larger damages, an epoxy patching compound is recommended. This type material is a two part, fast curing, easy sanding commercially available compound.

(5) Adhesion can be increased by roughing the bonding surface with sandpaper and by utilizing as much surface area for the bond as possible.

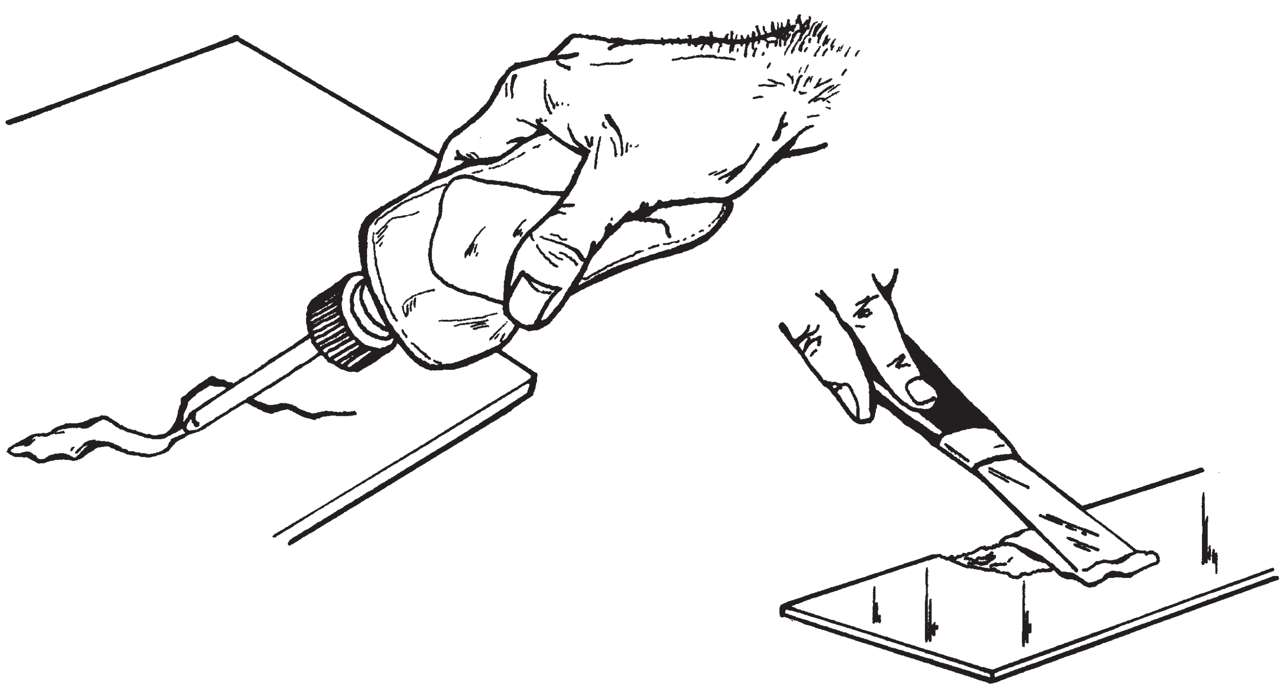
(6) The patching compound is mixed in equal portions on a hard flat surface using a figure eight motion. The damaged area is cleaned with perchlorethylene or V M & P Naphtha prior to applying the compound (see “Figure 3”).

(7) A mechanical sander can be used after the compound is cured, providing the sander is kept in constant motion to prevent heat buildup.

(8) For repairs in areas involving little or no shear stress, the hot melt adhesives, polyamids which are supplied in stick form, may be used. This type of repair has a low cohesive strength factor.

(9) For repairs in areas involving small holes, indentations or cracks in the material where high stress is apparent or where thin walled sections are used, the welding method is suggested.

871



Deep Scratches, Shallow Nicks and Small Holes
Figure 2

(10) This welding method requires a hot air gun and ABS rods. To weld, the gun should be held to direct the flow of hot air into the fusion (repair) zone, heating the damaged area and rod simultaneously. The gun should be moved continuously in a fanning motion to prevent discoloration of the material. Pressure must be maintained on the rod to ensure good adhesion (see “Figure 4”).

(11) After the repair is completed, sanding is allowed to obtain a surface finish of acceptable appearance.

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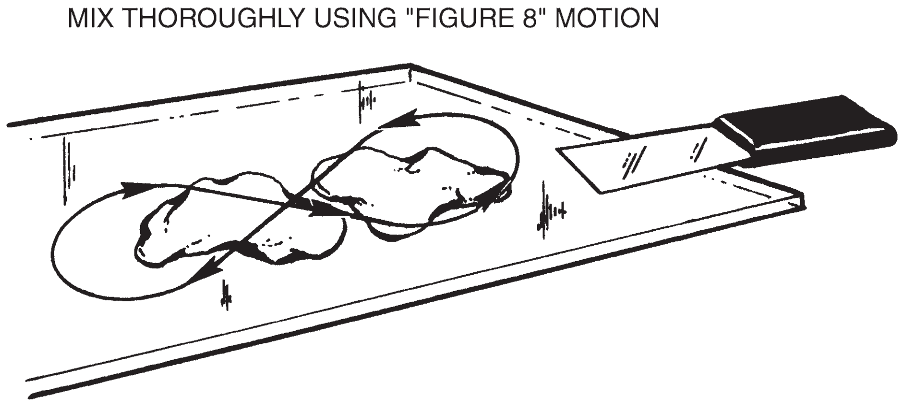

Mixing of Epoxy Patching Compound
Figure 3

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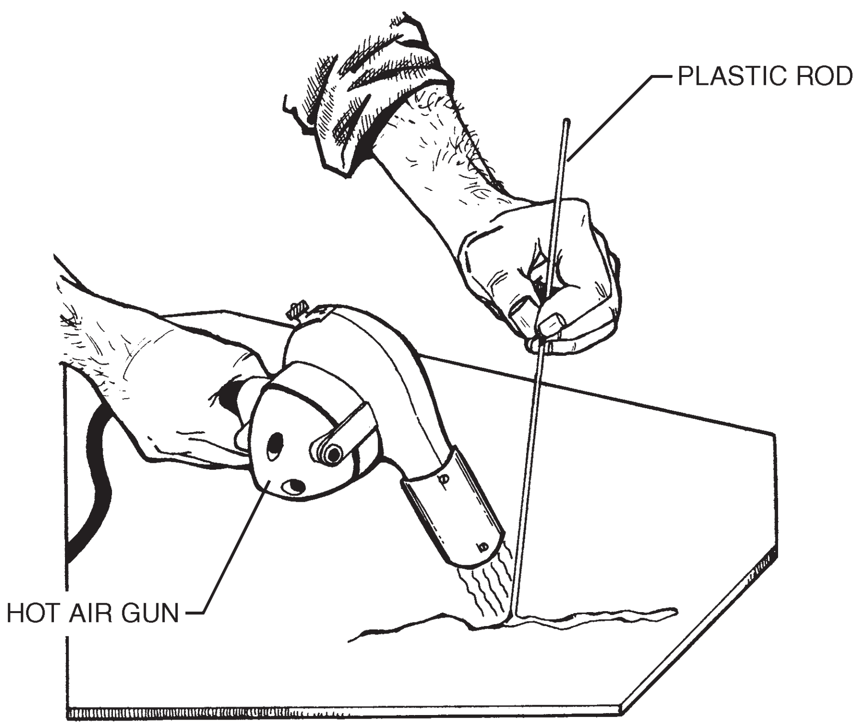

Welding Repair Method
Figure 4

D. Cracks

See “Figure 5”.

(1) Before repairing a crack in the thermoplastic part, first determine what caused the crack and alleviate that condition to prevent it from recurring after the repair is made.

(2) Drill small stop holes at each end of the crack.

(3) If possible, a double plate should be bonded to the reverse side of the crack to provide extra strength to the part.

(4) The crack should be “V” grooved and filled with repair material, such as solvent cement, hot melt adhesive, epoxy patching compound or it should be hot air welded, whichever is preferred.

(5) After the repair has cured, it may be sanded to match the surrounding finish.

E. Repairing Major Damage - Larger than 1 inch in diameter

See “Figure 6”.

(1) If possible, a patch should be made of the same material, and cut slightly larger than the section being repaired.

(2) When appearances are important, large holes, cracks, tears, etc, should be repaired by cutting out the damaged area and replacing it with a piece of similar material.

(3) When cutting away the damaged area, under cut the perimeter and maintain a smooth edge. The patch and/or plug should also have a smooth edge to ensure a good fit.

(4) Coat the patch with solvent adhesive and firmly attach it over the damaged area.

(5) Let the patch dry for approximately one hour before any additional work is performed.

(6) The hole, etc., is then filled with the repair material. A slight overfill of the repair material is suggested to allow for sanding and finishing after the repair has cured. If patching compound is used, the repair should be made in layers, not exceeding a 1/2 inch thickness at a time. This will allow the compound to cure and ensure a good solid buildup of successive layers as required.

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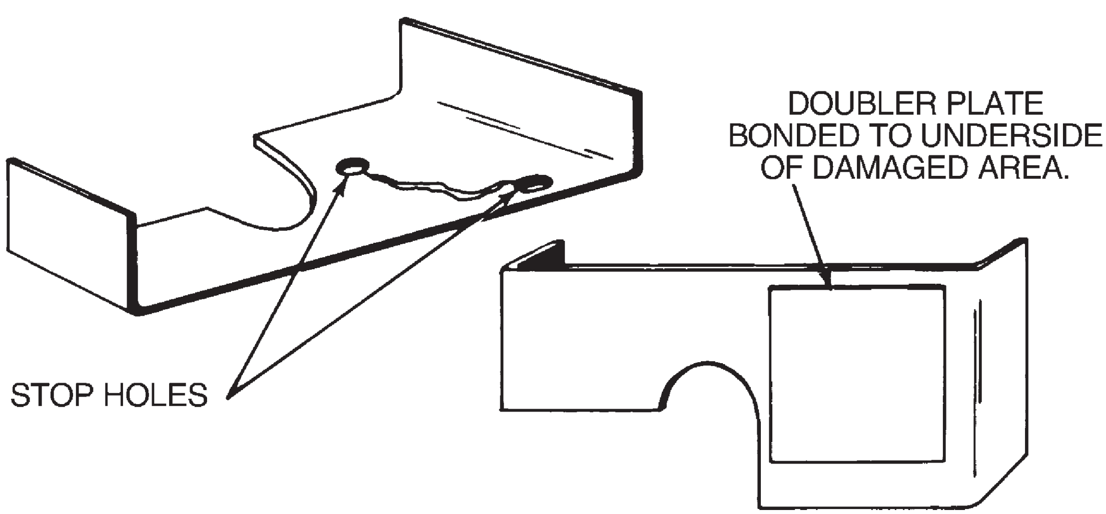
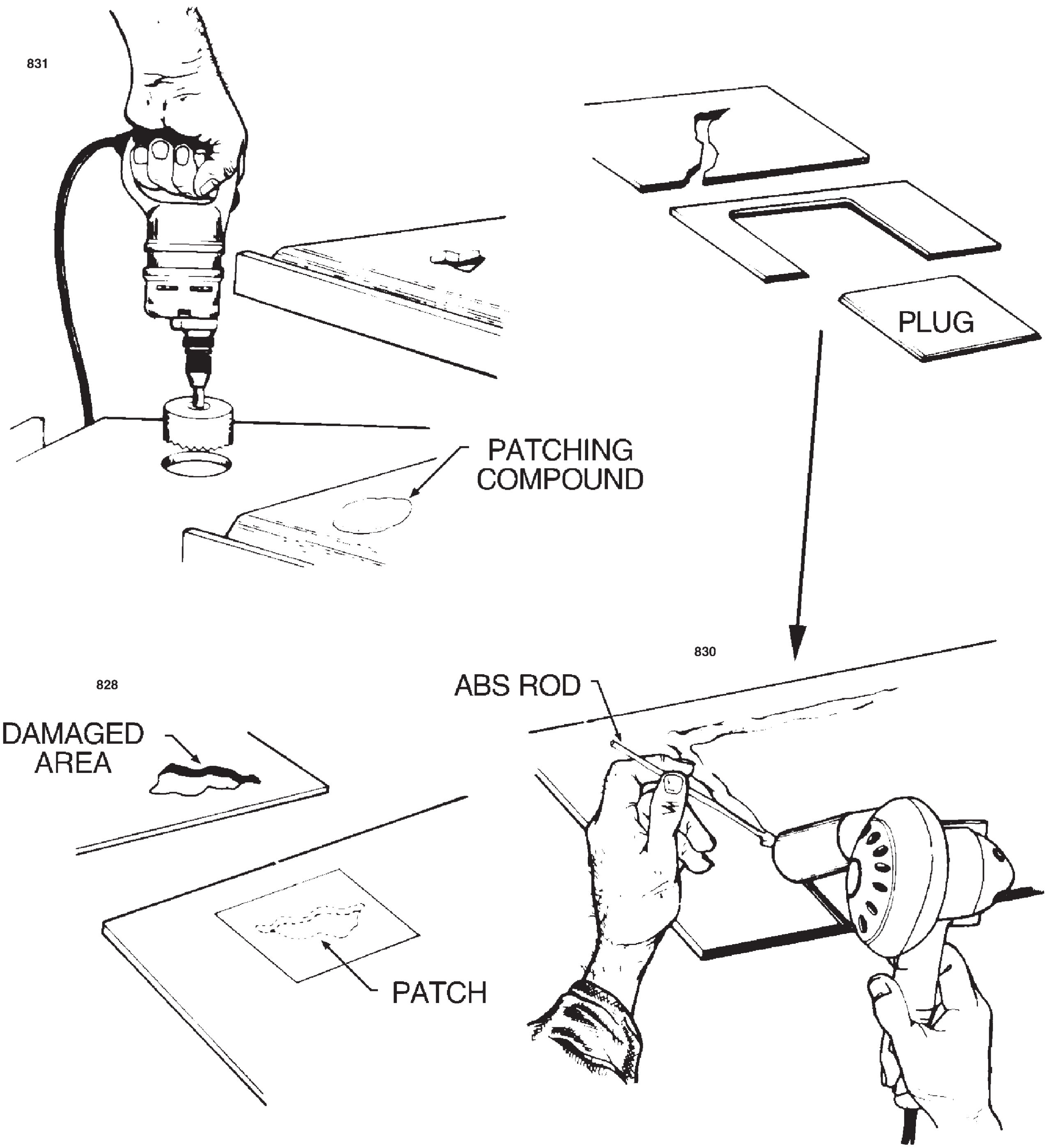


Repairing of Cracks
Figure 5



Various Repairs
Figure 6

F. Stress Lines

See "Figure 7".

(1) Stress lines produce a whitened appearance in a localized area and generally emanate from the severe bending or impacting of the material (see Figure 8).

(2) To restore the material to its original condition and color, use a hot air gun or similar heating device and carefully apply heat to the affected area. Do not overheat the material.

G. Painting the Repair

(1) An important factor in obtaining a quality paint finish is the proper preparation of the repair and surrounding area before applying any paint.

(2) It is recommended that parts be cleaned prior to painting, using a commercial cleaner or a solution made from one-fourth cup of detergent mixed with one gallon of water.

(3) The paint used for coating thermoplastic can be either lacquers or enamels, depending on which is preferred by the repair facility or customer.

NOTE: It is extremely important that solvent formulations be considered when selecting a paint, because not all lacquers or enamels can be used satisfactorily on thermoplastic. Some solvents used in the paints can significantly affect and degrade the plastic properties

(4) Another important matter to consider is that hard, brittle coatings that are usually best for abrasion resistance should not be used in areas which incur high stress, flexing or impact. Such coatings may crack, thus creating a weak area.

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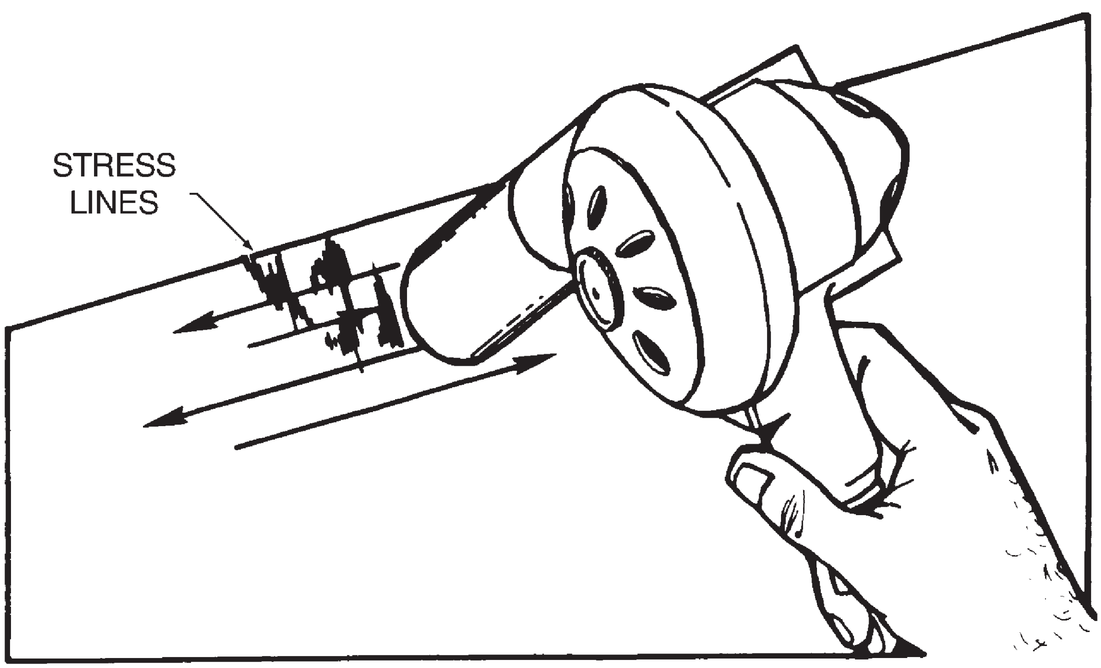
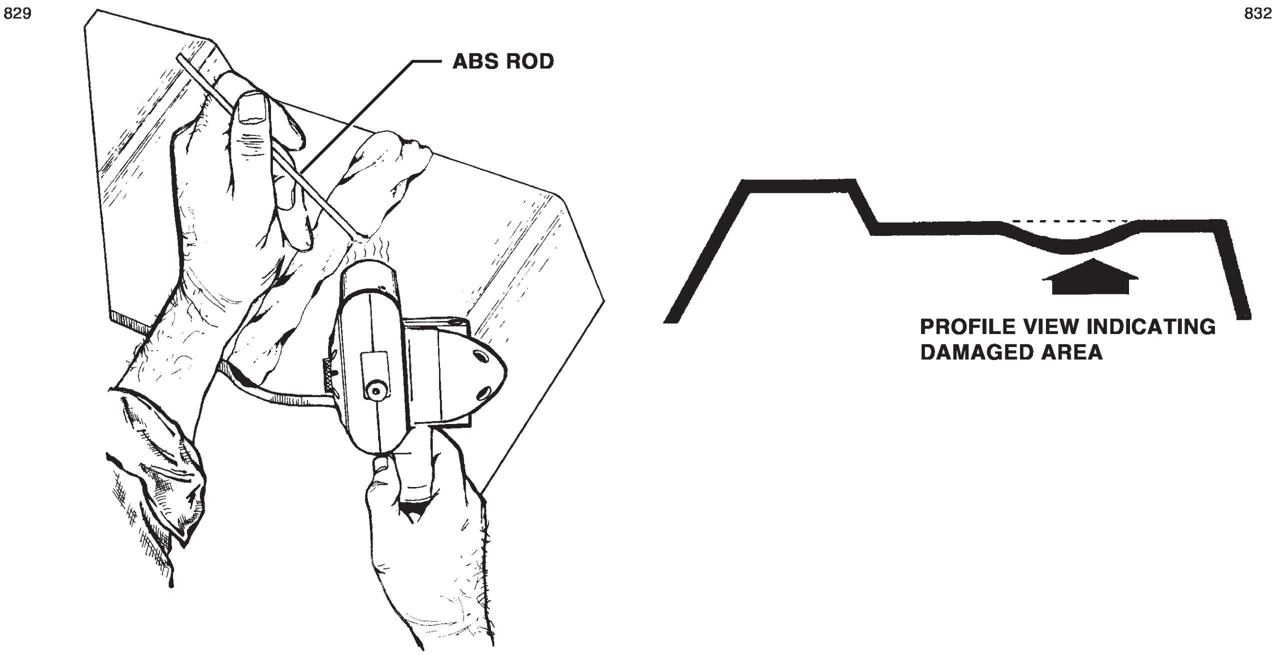


Repair of Stress Lines
Figure 7



Repair of Impacted Damage
Figure 8

3. Pressure Sensitive Safety Walk Installation

(PIR-PPS45010, Rev. G.)

Each of the steps on the lower cabin door have an 11.75 in. x 7.50 in. piece of pressure-sensitive safety walk installed on them.

A. Surface Preparation

(1) Allow newly painted surfaces to dry a minimum of 2 hours prior to applying the safety walk.

(2) If the paint becomes contaminated, remove the contamination with clean, dry rags or paper wipers moistened with a suitable solvent.

(3) Prior to applying the safety walk, scuff the applicable surface with Scotch-Brite and then wipe with a clean, dry cloth, ensuring that no moisture remains on the surface.

B. Application Procedure

(1) Do not apply when surface temperature is below 50°.

(2) Peel back the full width of the protective liner leading edge approximately two inches.

(3) Adhere this edge of the safety walk to the inboard edge of the step.

(4) Remove the remaining protective lines as the safety walk is being adhered to the step.

(5) Roll firmly with a long-handled, cylindrical brush, in both lengthwise directions.

4. Metal / Wire Stitching Repair

(PIR-PPS20024, Rev. A.)

CAUTION: METAL/WIRE STITCHING (AND THE ALTERNATE METHOD OF JOINING DESCRIBED BELOW) SHALL ONLY BE USED FOR NON-STRUCTURAL, NON-LOAD CARRYING APPLICATIONS.

A metal/wire stitching process is used to staple fabric and rubber seal materials to engine baffles and some composite materials. The following alternate method of joining is approved for field use when replacing these fabric and rubber seal materials.

A. Alternate (Rivet) Method of Joining.

(1) Substitute two rivets in lieu of each staple where stitching was previously used or is specified. Maintain a minimum of .75 inch spacing between rivets.

(2) When materials being joined include Stainless Steel, Galvanized Steel, or Steel, use:

(a) MS20615M4 Rivet (Monel) and NAS1149CN432R Washer (See “Figure 9”.)

(b) Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.

(3) When materials being joined include only aluminum and non-metallic materials, use:

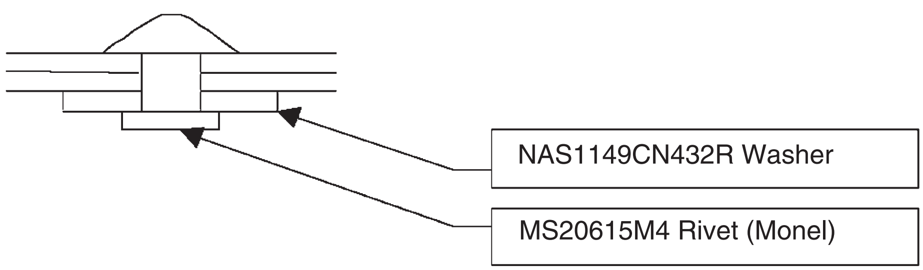
(a) MS20470A4 Rivet and NAS1149DN432H Washer (See “Figure 9”.)

(b) Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.

When materials being joined include Stainless Steel, Galvanized Steel or Steel, use:

MS20615M4 Rivet (Monel)
NAS1149CN432R Washer

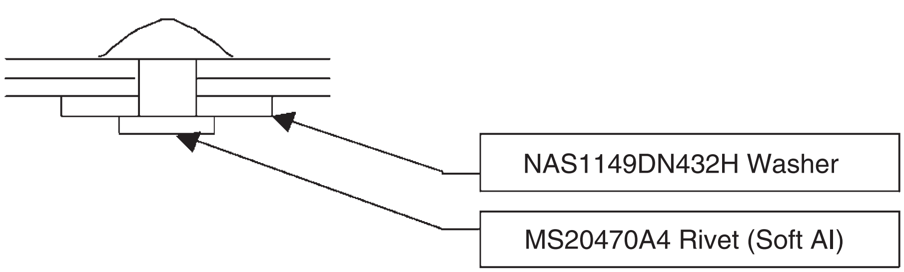
Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.



When materials being joined include only aluminum and nonmetallic materials use:

MS20470A4 Rivet
NAS1149DN432H Washer

Install with manufactured (factory) head against hardest material. Install washer against opposite side of joint and upset rivet (bucktail) against washer.



Metal / Wire Stitching Repair
Figure 9

ELECTRICAL BONDING

1. General

(PIR-PPS55006, Rev. AI.)

See also 23-60-00.

All electrical and electronic equipment and specified components shall be installed in such a manner as to provide a continuous, low-resistance path (bonds) from the equipment enclosure/component to the airplane structure. Bonds must be installed to ensure that the structure and equipment are electrically stable and free from the hazards of lightning, static discharge, electrical shock, etc.

A. All parts shall be bonded with as short a lead as possible.

B. All bonding surfaces shall be cleaned prior to the installation of the bonded joint.

C. All nuts used in bonding shall be of the self-locking type. (Do Not use fiber-locking type).

D. All electrical bonding shall be accomplished without affecting the structural integrity of the airframe.

2. Inspections

(PIR-AC 43.13-1, Rev. B.)

A. 100 Hour

Each 100 hours, visually inspect shield and shield terminations of each electrical harness for integrity, condition, and security. If electrical arcing is evident, check for intermittent contact between conducting surfaces. Arcing can be prevented by bonding or insulation, as appropriate.

(1) Inspect the components listed in “Chart 1” as follows:

(a) Bond connections shall be secure and free from corrosion.

(b) Bonding jumpers installed so as not to interfere in any way with the operation of moveable components of the aircraft.

(c) No self-tapping screws used for bonding purposes.

(d) Exposed conducting frames or parts of electrical or electronic equipment should have a low-resistance bond of less than 2.5 milliohms to structure. If the equipment design includes a ground terminal or pin, which is internally connected to such exposed parts, a ground wire connection to such terminal will satisfy this requirement.

(e) Parts shall be bonded directly to the primary structure rather than to other bonded parts.

(f) Where aluminum or copper is bonded to dissimilar metallic structures, ensure installed hardware (typically washers) is as called out in the parts catalog to minimize electrolytic corrosion and ensure the hardware should corrode first.

(2) In PA-46R-350T’s and PA-46-350P’s (S/N’s 4636375 and up), inspect the Lightning Diverter (see Figure 2) bond between the feed through (i.e., the grounding screw) and the airframe structure. Bonding resistance must be 0.01 ohm or less.

B. On Condition

Whenever any electrically bonded component (see “Chart 1”) is removed and reinstalled, or visual inspection reveals the electrical bonding to be suspect, measure resistance between component and aircraft structure.

To ensure proper operation and suppression of radio interference from hazards, electrical bonding of equipment must not exceed the maximum allowable resistance values specified in “Chart 1”.

(1) Measurements should be performed after the grounding and bonding mechanical connections are complete to determine if the measured resistance values meet the basic requirements.

(2) A high quality test instrument (an AN/USM-21A or equivalent) will accurately measure the very low resistance values specified.

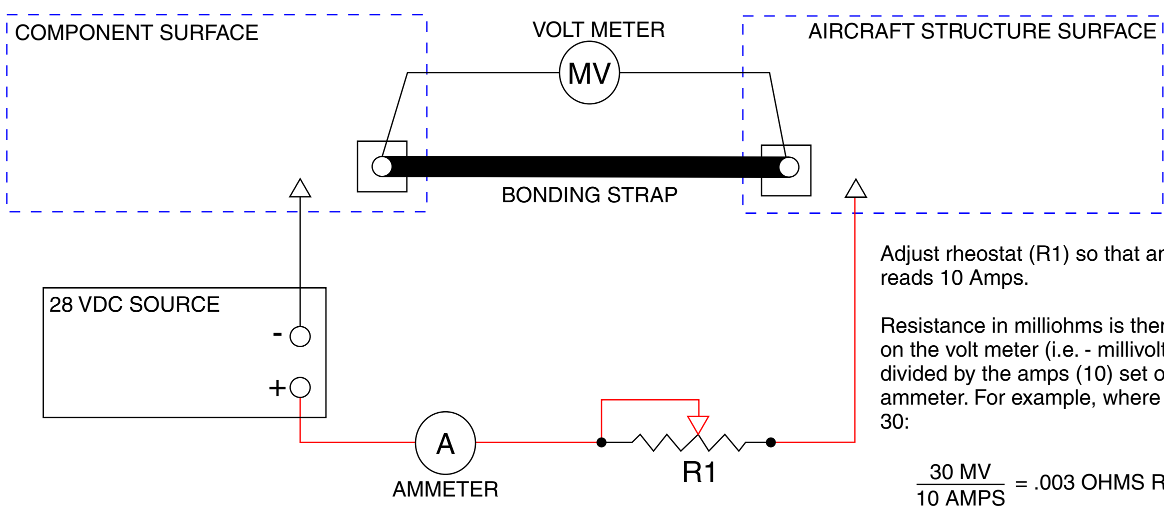
(3) Another method of measurement is the millivolt drop test as shown in “Figure 1”.

CHART 1
ELECTRICAL BONDING RESISTANCE INDEX

Component	Maximum Allowable Resistance Value in Ohms
Engine Mount(s)	.003
Generator(s)	.010
Ailerons	.003
Elevator / Stabilator	.003
Rudder	.003
Alternator(s)	.010
Trim Tab(s)	
Conventional Hinge	.003
Piano Hinge	.010
Instrument Panel Inserts	.010
Exterior Lights Mounted on Non-Conductive Material	.003
Avionics 'Black Boxes'	.003
NOTE: Harnesses should be installed and connected for this check, internal chassis wiring through the connector to ground is permissible for this grounding.	
Battery Ground Point	.010
Static wick mounting plates (TCO Model B-4) P/N 452-094	1.00

NOTE: Where jumper wires or cables are used to accomplish a proper bond, resistance between the jumper terminal and the component or structure shall not exceed .001 ohms. The controlling points for measuring resistance will be within the limits of the cleaned area to be bonded and within 1/4 inch of the exterior limits of the bonding jumper terminal or material called for in the bill of materials of the drawing.

NOTE: Resistance to ground will be measured from wire terminal to structure for electrical / electronic equipment not internally grounded and from mounting flange to structure for equipment that is internally grounded.



Adjust rheostat (R1) so that ammeter (A) reads 10 Amps.

Resistance in milliohms is then the reading on the volt meter (i.e. - millivolts (MV)) divided by the amps (10) set on the ammeter. For example, where MV equals 30:

$$\frac{30\ \text{MV}}{10\ \text{AMPS}} = .003\ \text{OHMS RESISTANCE}$$

Millivolt Drop Test
Figure 1

3. Lightning Diverter

(PIR-PPS50075, Rev. H.)

In all PA-46R-350T's and PA-46-350P's S/N's 4636375 and up, a Lightning Diverter is installed on each side of the dorsal fairing (see "Figure 2" on page 51804).

A. Inspection

Each 100 hours time-in-service, inspect the Lightning Diverters as follows:

NOTE: Any of the following conditions is cause for replacement.

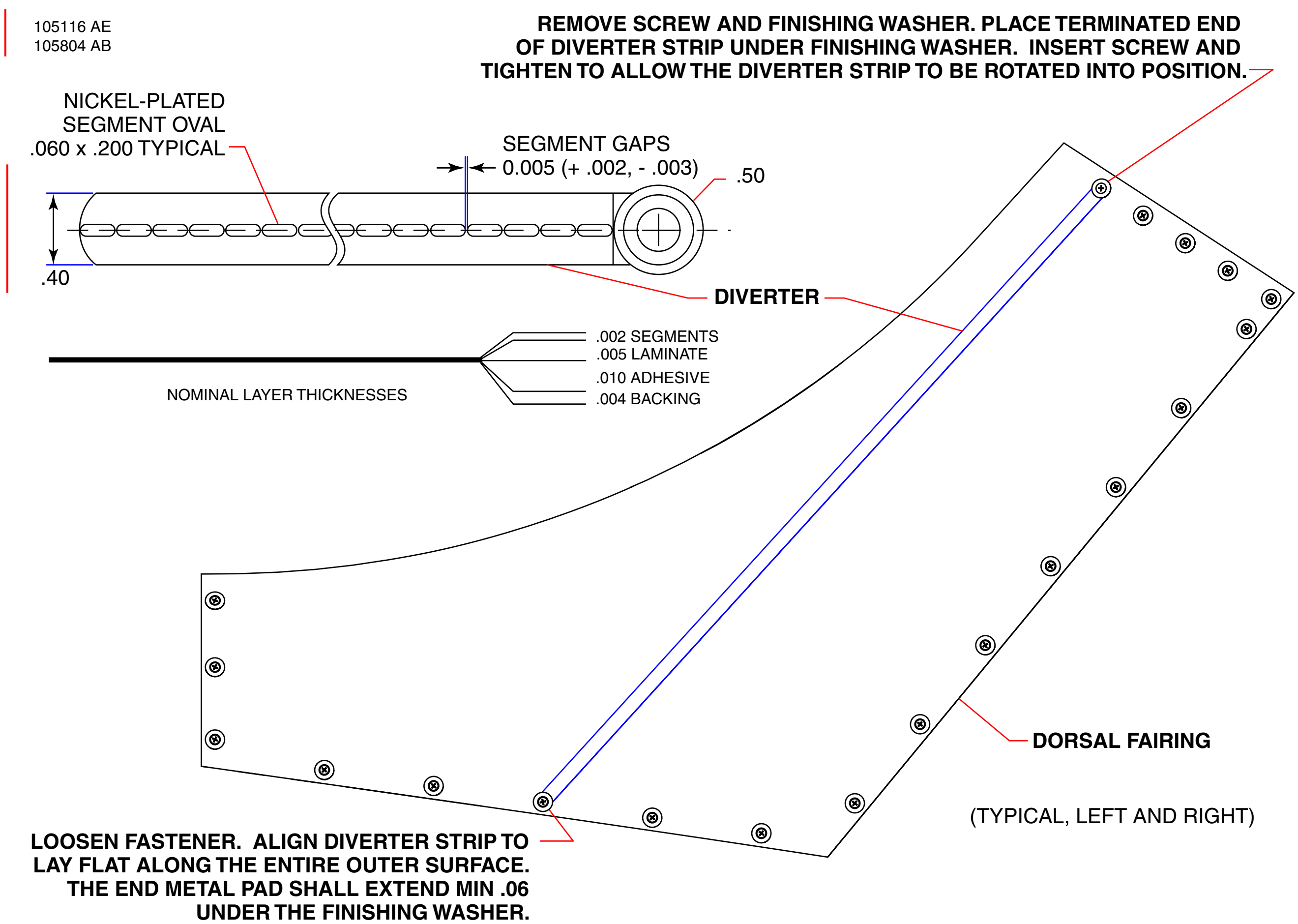
(1) Paint or adhesive covering the Nickel-plated segments (conductive segments).
(2) More than 10% of the Nickel-plated segments (conductive segments) missing.
(3) More than three adjacent Nickel-plated segments (conductive segments) missing.
(4) Delamination of the assembly.
(5) Any breaks, cuts, cracks, or tears extending beyond one layer.
(6) Any break, cut, crack, or tear that causes a disconnection of the Nickel-plated segments from the diverter strip.

B. Replacement

Order and install Piper Kit No. 88559-002 or 88559-003, as appropriate, per "Chart 2". Each kit provides the instructions and parts to replace one lightning diverter strip. (Each airplane has two diverter strips.)

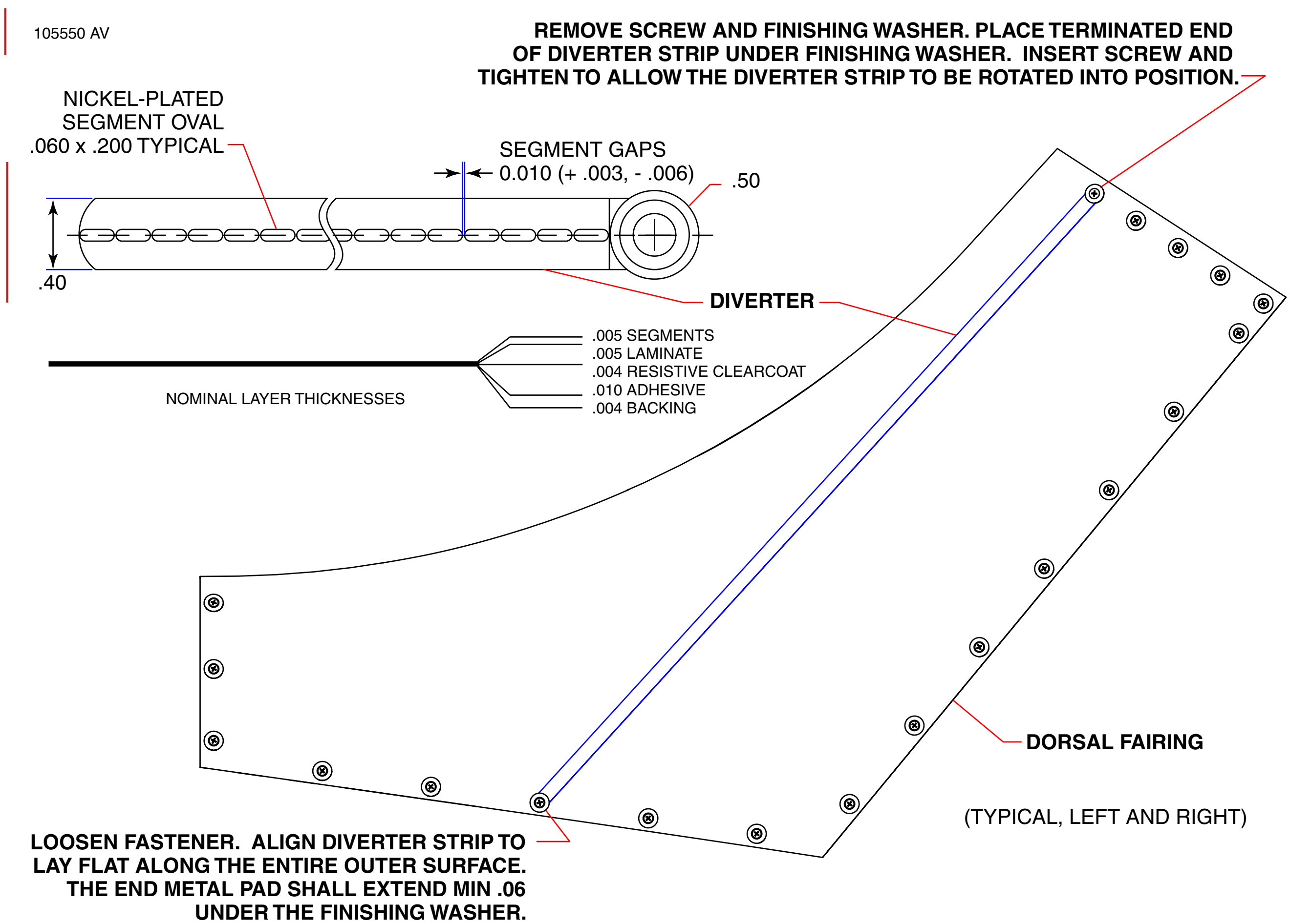
CHART 2
SERVICE KIT APPLICABILITY

Airplane Model	Serial Numbers	Kit Number
PA-46-350P Mirage/M350	4636375 thru 4636459, 4636461, 4636462, and 4636481	88559-002
	4636460, 4636463 thru 4636480, 4636482 and up	88559-003
PA-46R-350T Matrix	4692001 thru 4692133, 4692141, 4692149, and 4692153	88559-002
	4692134 thru 4692140, 4692142 thru 4692148, 4692150 thru 4692152, 4692154 through 4692214	88559-003
NOTE: Instructions for Kit P/N 88559-002 and 88559-003, Kit – Diverter Strip Replacement, must be marked Revision C or later.		



Effectivity
4636375 and up
4692001 and up

Lightning Diverter Installation
Figure 2 (Sheet 1 of 2)



Lightning Diverter Installation
Figure 2 (Sheet 2 of 2)

Effectivity
with Garmin G1000

C. Repair

Limited repairs to loose or lifting edges may be accomplished as follows:

(1) Carefully mask an area on the dorsal fairing 1/16 inch around the diverter strip and on the diverter strip 1/16 inch from the edge.

(2) Solvent clean the area to be abraded and the area under the loose section of the diverter strip using isopropyl alcohol. Avoid prying the diverter strip up so as not to deform the area.

(3) Prepare the exposed edge area of the diverter strip by abrading with 220 or finer abrasive paper.

(4) Repeat the solvent cleaning as directed above.

(5) Apply two more layers of masking tape on top for the previously applied masking tape being careful to align the edges of the tape because these will define the final repair shape.

(6) Shake 3M Primer 94 well before opening. Wipe faying surfaces in repair area with a pre-saturated 3M Primer 94 wiper using the minimum amount that will coat the repair areas. Allow surfaces to dry for 10 minutes (minimum) and 30 minutes (maximum) before applying repair adhesive.

(7) Apply 3M DP-100 (P/N 279-471) under the loose diverter strip and seat the strip.

(8) Apply the 3M DP-100 over the repair area flush with the top of the masking tape, using a straight edge to create a continuous even overlay of the repair area.

(9) As soon as the surface of the repair is tack free remove the masking tape by pulling away from the repair and being careful not to disturb the repair.

(10) Inspect the repair for lifted ragged edges. If found, wait 24 hours and if they are still present, trim as required to a fair condition and apply a thin layer of 3M DP-100 to the trimmed area to form a smooth transition to the adjacent surface while staying as close as possible to edge of the sealant.

(11) Allow the repair to cure for a minimum of 24 hours before returning to service.

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4. G1000/G1000 NXi Equipment Electrical Bonding Check

(PIR-PPS55026, Rev. H; 107977 F; 107998 New.)

The following procedures verifies proper electrical bonding for G1000/G1000 NXi system equipment.

NOTE: Any area that fails shall be re-bonded and re-tested per AC 43.13-1 latest revision.

A. Required Equipment

A milli-ohmmeter or equivalent.

B. GRS XX AHRS (2 ea.)

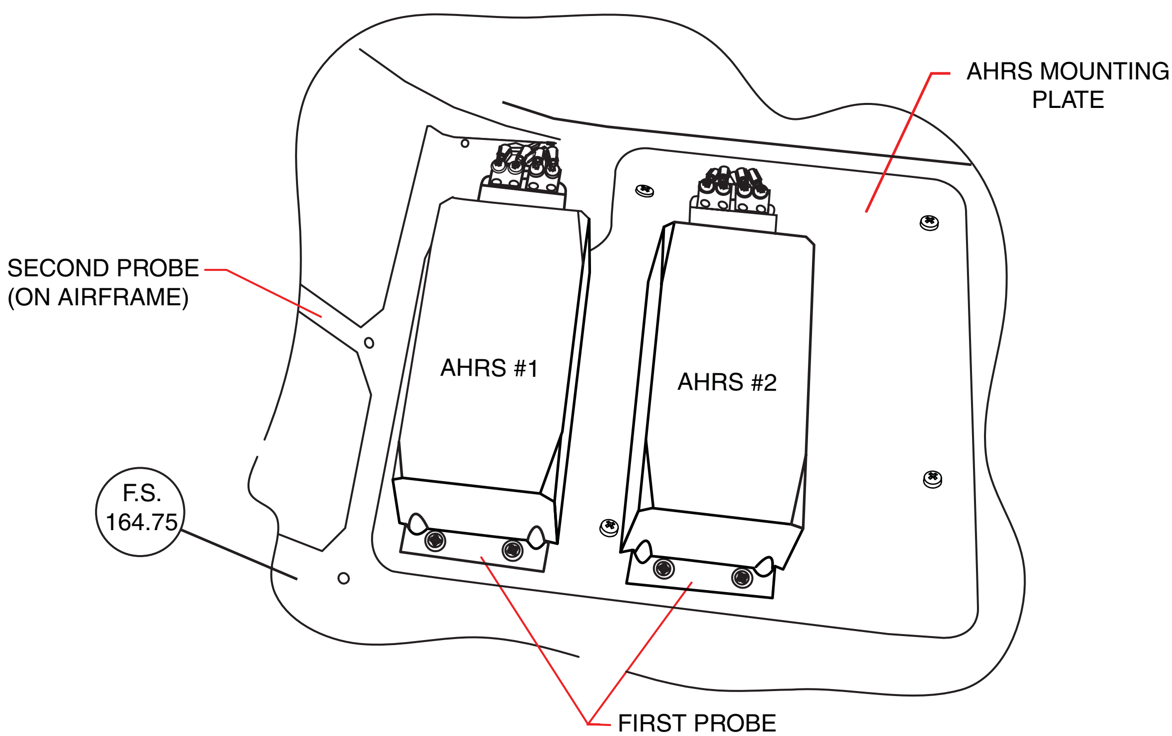
Refer to “Figure 3”.

(1) Test each unit between the two points shown.
(2) Make sure the harness to the units are not connected or that the units are not removed from the rack for this test.
(3) PASS: = or < 2.5 mohms

C. GPS Antennas (2 ea.)

Refer to “Figure 4” on page 51809.

(1) Test from the screws to a point on the airframe that is two feet away,
(2) The coax cable must be removed for this test.
(3) PASS: = or < 2.5 mohms



GRS XX Electrical Bonding Check
Figure 3

D. GMU 44 Magnetometer (2 ea.)

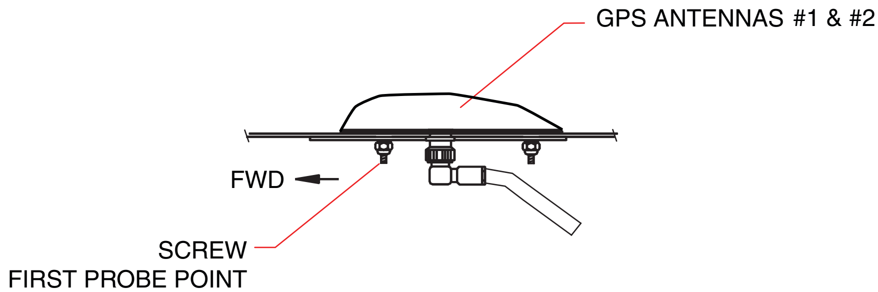
Refer to “Figure 5”.

(1) Test between the two points shown.

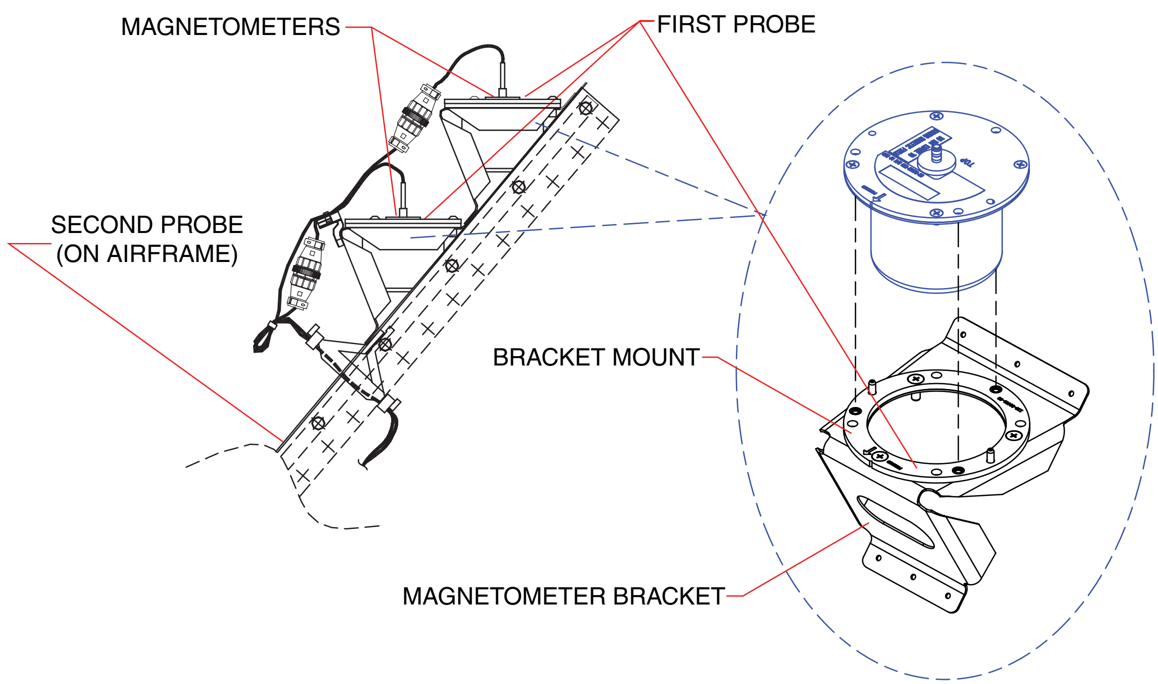
(2) This test is to verify that the bracket mount is bonded to the bracket and to the airframe.

(3) Make sure the unit is unplugged or removed from bracket for this test.

(4) PASS: = or < 2.5 mohms.



GPS Antenna Electrical Bonding Check
Figure 4



Magnetometer Electrical Bonding Check
Figure 5

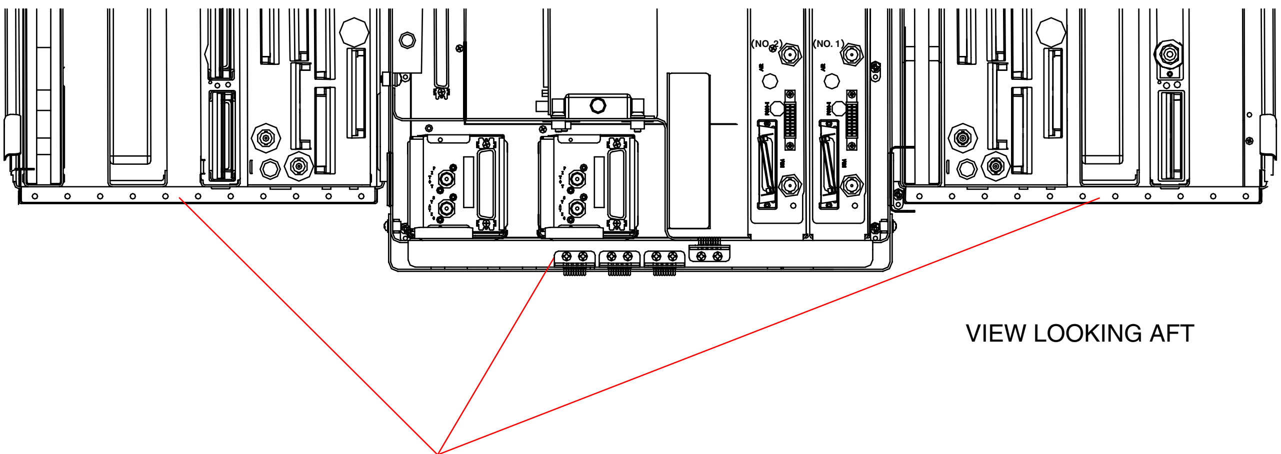
E. MFD and PFD Racks

Refer to “Figure 6”.

(1) Test between the rack and aircraft floor.

(2) This is the main rack that holds all of the other equipment.

(3) PASS: = or < 10 mohms



MFD and PFD Racks Electrical Bonding Check
Figure 6

F. MFD (1 ea.) and PFD (2 ea.)

Refer to”Figure 7” on page 518011.

(1) Test on a clean metal spot on the aft side of GDU to the aircraft floor.

(2) The units must be installed for this test.

(3) “Figure 7” shows the front of the panels only as a reference to show an approximate location to test.

(4) PASS: = or < 20 mohms.

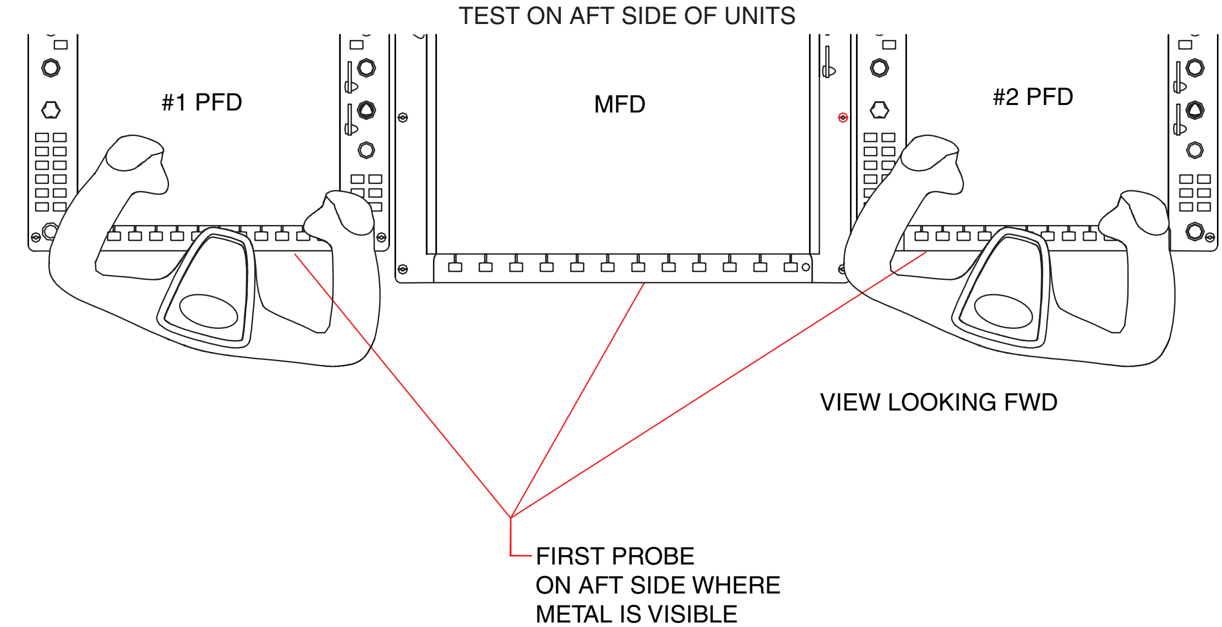
G. OAT Probes (2 ea.)

Refer to “Figure 8” on page 518011.

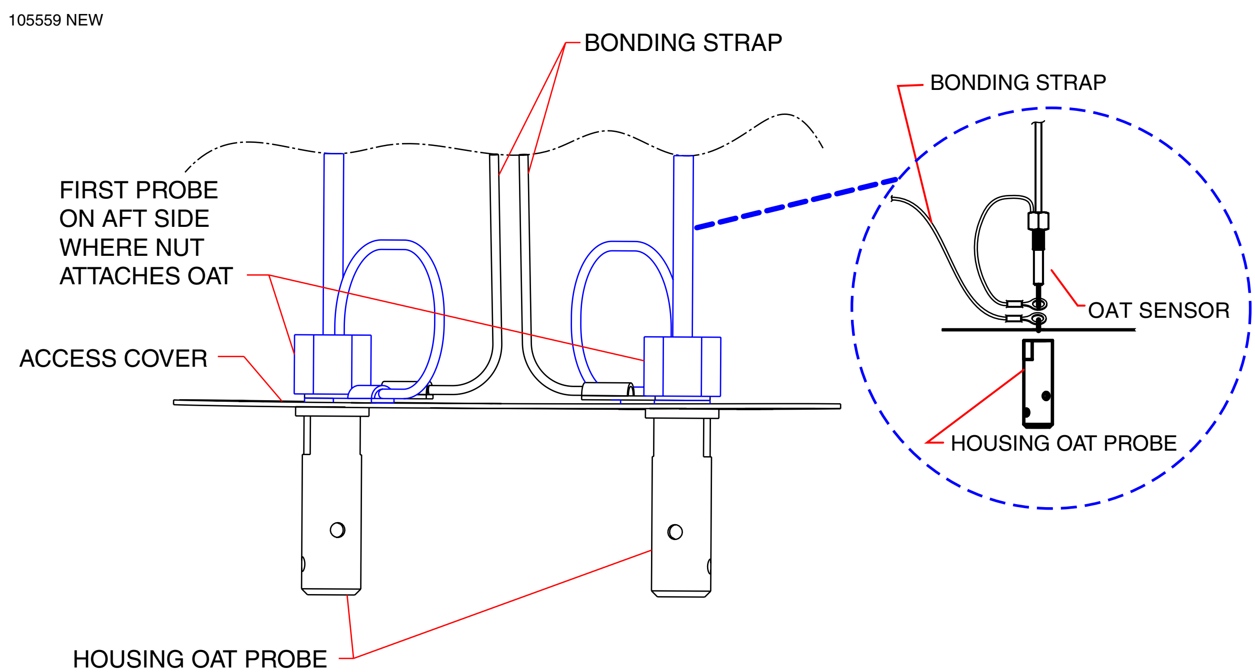
(1) Test between the OAT probe and a wing frame approximately Two feet away.

(2) Use the nut on the OAT probes as test points.

(3) PASS: = or < 10 mohms



MFD and PFD's Electrical Bonding Check
Figure 7



OAT Probes Electrical Bonding Check
Figure 8

H. GSA 8X/GSM 8X Servo (4 ea.)

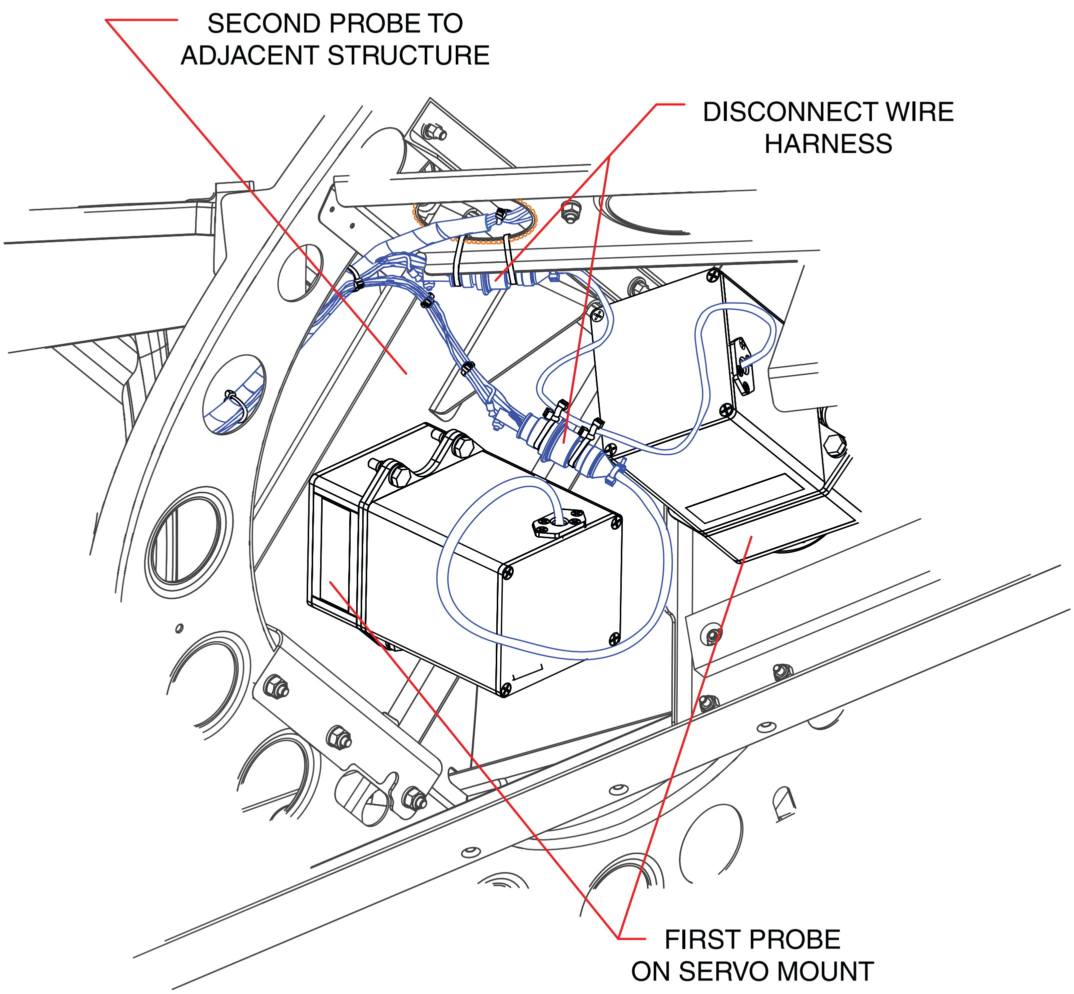
Refer to “Figure 9”.

(1) Disconnect wire harness from servo.

(2) Test between servo mount and adjacent structure.

(3) PASS: = or < 2.5 mohms.

(4) After passing test, reconnect wire harness.



TYPICAL ON ALL 4 SERVOS

Servo Electrical Bonding Check (GSA 81/GSM 85(A) shown)
Figure 9

I. GDL 69A / GDL 69eA

If installed.

(1) Disconnected wiring harness and antenna cable.

NOTE: To accomplish this, the backplate assembly with attached connectors can be removed from the GDL mounting rack.

(2) Measure the bonding resistance from the rack to airframe ground.

(3) PASS: = or < 10 mohms.

(4) After passing test, reconnect wiring harness and antenna cable.

J. XM Antenna

If installed.

(1) Disconnected antenna cable.

(2) Measure antenna bonding from the XM connector (TNC external ground) on the antenna to a nearby exposed portion of conductive aircraft structure (example: exposed rivet on fuselage stringer).

(3) PASS: = or < 10 mohms.

(4) After passing test, reconnect antenna cable.

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CHAPTER

52

DOORS

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CHAPTER 52

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
52-List of Effective Pages	1	Nov 30/17			
	2	Sep 15/09			
52-Table of Contents	1	Jan 15/13			
	2	Nov 30/17			
52-10-00	1	Feb 15/14			
	2	Jan 15/13			
	3	Jan 15/13			
	4	Jan 15/13			
	5	Jan 15/13			
	6	Jan 15/13			
	7	Jan 15/13			
	8	Jan 15/13			
	9	Jan 15/13			
	10	Jan 15/13			
	11	Jan 15/13			
	12	Jan 15/13			
	13	Jan 15/13			
	14	Jan 15/13			
	15	Jan 15/13			
	16	Jan 15/13			
52-20-00	1	Jul 1/10			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Sep 15/09			
52-30-00	1	Sep 15/09			
	2	Sep 15/09			
52-70-00	1	Nov 30/17			
	2	Nov 30/17			

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CHAPTER 52 - DOORS

TABLE OF CONTENTS

CHAPTER 52 - DOORS

TABLE OF CONTENTS (continued)

PASSENGER / CREW

1. Description

This airplane is provided with a cabin door on the left side of the fuselage, between the passenger seats. The cabin door is in two sections, the upper door having a window and the lower door having two boarding steps that assume their proper position when the door is lowered. Signal flags are incorporated in each of the cabin door's locking pins, to show each pin's position.

The emergency exit is on the right-hand side of the fuselage at the center seat. See 52-20-00.

NOTE: See 51-70-00 for instructions on installing the pressure-sensitive safety walk strips on the lower cabin door steps.

2. Cabin Door

A. Upper Half

(1) Removal

(a) Rotate the handle to the unlocked position and open the door.

(b) Open the lower door.

(c) Disengage the door holder assembly from the door.

(d) Hold the door partly open and relieve the weight of the door from the hinge pin to facilitate hinge pin removal.

(e) Remove the hinge pin and lift off the door.

(2) Installation

(a) Align the door hinge with the fuselage hinge and insert the hinge pin.

(b) Attach the door holder assembly to the door.

B. Lower Half

(1) Removal

(a) Open both upper and lower doors to full open position.

(b) (PA-46-350P) Clamp both cables with small vice grips at the point where each cable exits the cable reel mechanism in the door half. Disconnect the cable upper end from the fuselage.

(c) (PA-46R-350T) Remove the bolts, washers and nuts which secure the lower end of the cable assembly to the lower door.

(d) Relieve the weight of the door from the hinge and remove the hinge pin.

(e) Lift the door off of the lower hinge.

(2) Installation

(a) Align the door hinge with the fuselage hinge and insert the hinge pin.

(b) (PA-46-350P) Connect the cable upper end to the fuselage. Remove the small vice grip clamping each cable. Raise and lower the door half, verifying that the cable reel mechanism works smoothly.

(c) (PA-46R-350T) Attach the cable assembly to the door, using the previously removed bolts, washers, and nuts.

3. Door Support

A. Upper Door Half

The upper door half is held open by a hydraulic door support, located at the door's upper left-hand corner. The unit is designed to provide 65 to 80 pounds of pressure when fully extended.

(1) Removal

(a) Open the upper door. Relieve door weight on the door support.

(b) Support the door and remove the retaining nuts and washers from both ends of the door support.

(c) Remove the door support.

(2) Installation

(a) Position the door support on the door and fuselage.

(b) Secure the door support to the door and to the door frame, using the previously removed nuts and washers.

B. Lower Door Half (PA-46-350P)

Three different door support systems may be installed. Later airplanes all use the simple two-piece cable assemblies. Earlier airplanes use one of two spring-loaded cable reel systems.

(1) Set-up

Open both upper and lower door halves. Support the lower door half to relieve the weight of the door from the hinge and the cable assemblies.

(2) S/N's 4636446 and up

(a) Removal

1) Complete set-up, above.

2) Remove the cable retaining bolts, washers, and nuts from the door lug and door frame lug.

3) Remove the cable assembly.

(b) Installation

1) Attach the eye end of the cable to the door frame lug.

2) Attach the yoke end of the cable to the door lug.

(3) S/N's 4636001 thru 4636445 as built

NOTE: Always replace and rig one cable at a time.

(a) Removal

1) Complete set-up, above.

2) Remove the lower cabin door upholstery/covers to expose the cable/spool assemblies.

3) Lower the door and mark the spool for counting turns.

4) Grab the cable and hold against the spring tension (vise grips help here) and disconnect the cable from the fuselage.

5) Carefully allow the cable to retract while counting the total number of revolutions of the spool to relieve the spring tension.

NOTE: This will require removal of cable stops, routing through pulleys, etc.

6) Remove the spool to the bench.

(b) Cable Replacement

Drill out the rivet attaching the cable to the spool. Rivet the new cable in place.

(c) Installation

1) Wind the cable the same number of revolutions it took to relieve tension in step 4), under Removal, above.

NOTE: This will require winding the cable all the way, re-installing the spool, and then turning the cable/spool assembly to get the same number of turns. This will create a preload: lock the spool in position while routing the cable through the pulleys, guides etc. The default pre-load is one and a half turns.

NOTE: If replacing a broken cable, after completing Cable Replacement, above, wind the cable all the way, install the spool, and then preload the spring one and a half turns and lock the spool as above.

2) Reattach the cable at the fuselage. Let the door down, and apply gentle foot pressure on the bottom step.

3) Adjust the stop so that both cables bottom out at the exact same time.

4) Reinstall the upholstery/covers.

5) Helpful hints:

a) The small washer is under the rivet at the eyelet end.

b) If wound too tight; hold the reel with visegrips, pull out about a foot of cable, and remove one loop.

c) Install jamb nut on “T” handle all the way to the end.

d) Adjust cable plates by slotting. Make sure they are not installed backwards.

(4) S/N’s 4636001 thru 4636445 with Kit No. 88423-002 Installed

See “Figure 1”.

NOTE: Always replace and rig one cable at a time.

(a) Removal

1) Complete set-up, above.

2) Remove the lower cabin door upholstery/covers to expose the cable/spool assemblies.

3) Grab the cable and hold against the spring tension (vise grips help here) and disconnect the cable from the fuselage.

a) Forward Cable - Remove socket head cap screw, compression spring, and tumbler pin from handle. Then remove clevis pin and separate cable end from handle.

b) Aft Cable - Remove screw, washers, spacer, and bushing from nutplate on fuselage structure and remove cable end.

4) Carefully allow the cable to retract relieve the spring tension.

NOTE: This will require removal of cable stops, routing through/removal of pulleys, etc.

5) Remove the spool (and damper assembly, if aft spool) to the bench.

NOTE: Removing spool may be easier if the associated decal holder is removed first.

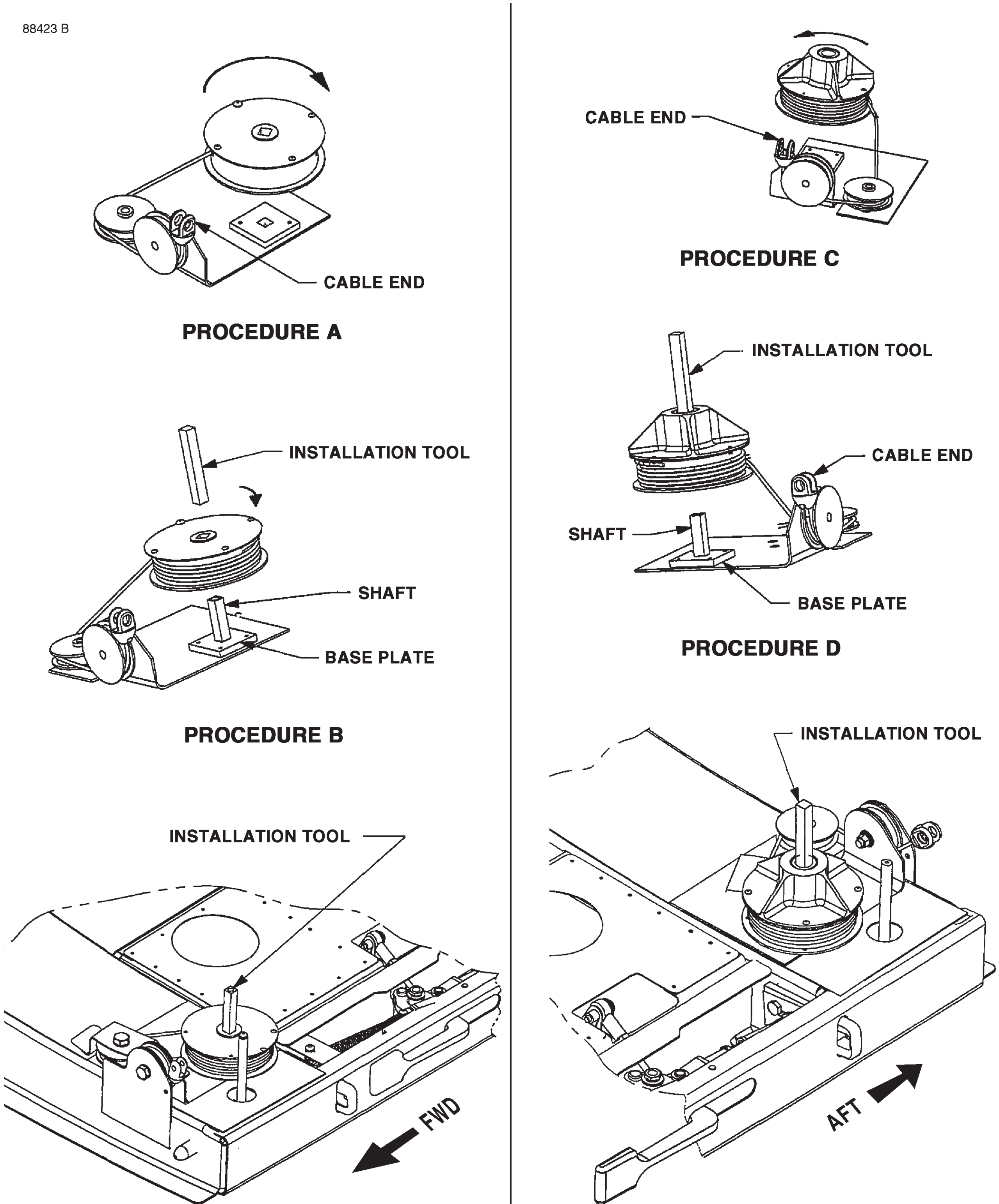
(b) Cable Replacement

1) Drill out the rivet attaching the cable to the spool.

2) Wet install with fluid resistant epoxy primer (Piper P/N 279-108 or equivilent) and rivet the new cable (106615-002 FWD and 106598-002 AFT) in place.

3) Grind material on cable terminal flush to match profile of spool.

NOTE: Alternatively, replace the entire cable spring motor. Replacement spring motors (106617-002 FWD and 106601-002 AFT) come with a new cable installed.

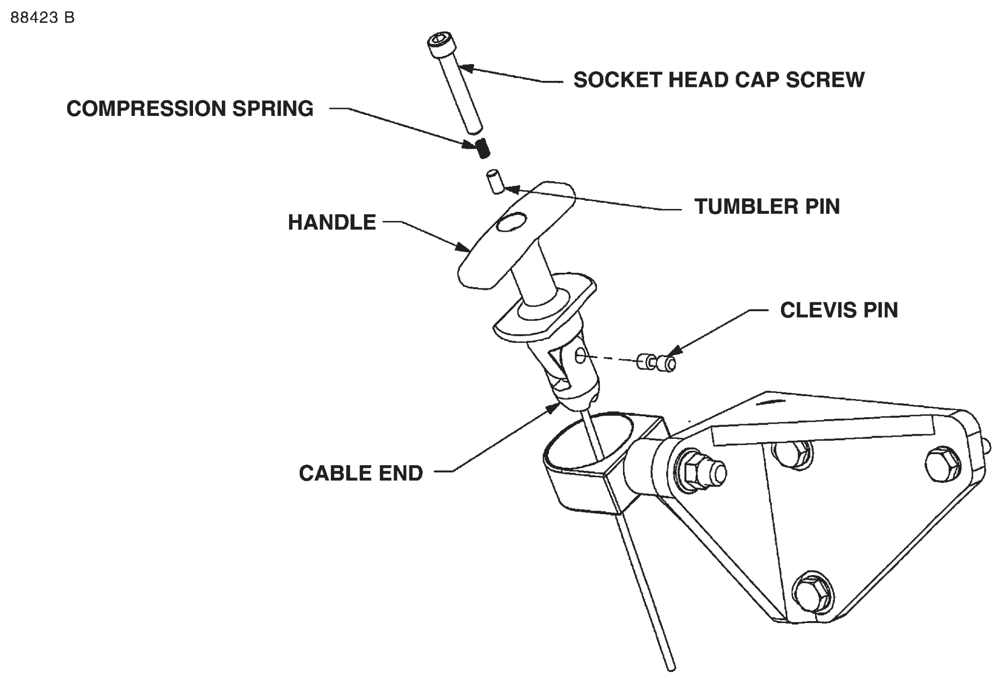


FORWARD CABLE SPOOL INSTALLATION

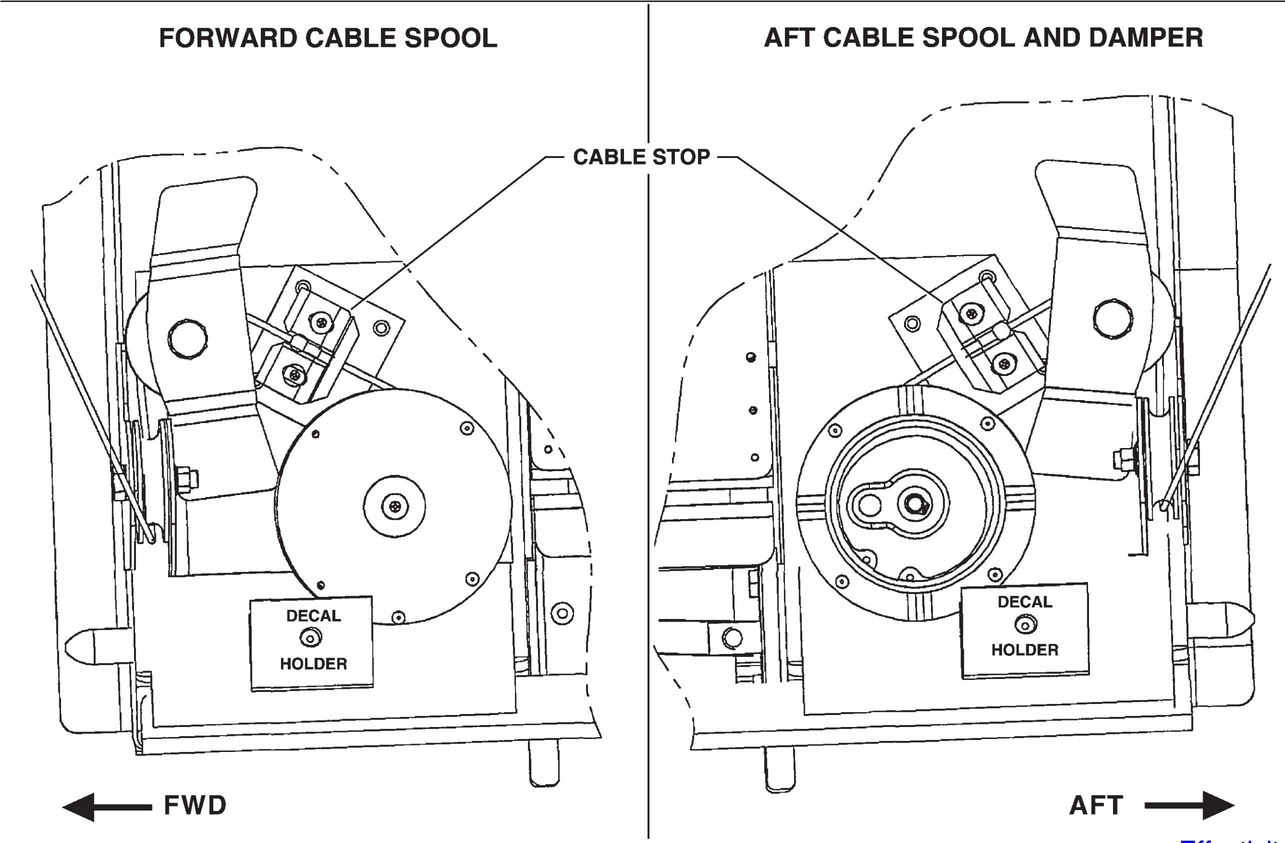
AFT CABLE SPOOL INSTALLATION
(DAMPER NOT YET INSTALLED)

Effectivity
4636001 thru 4636445
with Kit No. 88423 Installed

Lower Cabin Door Support Installation
Figure 1 (Sheet 1 of 3)

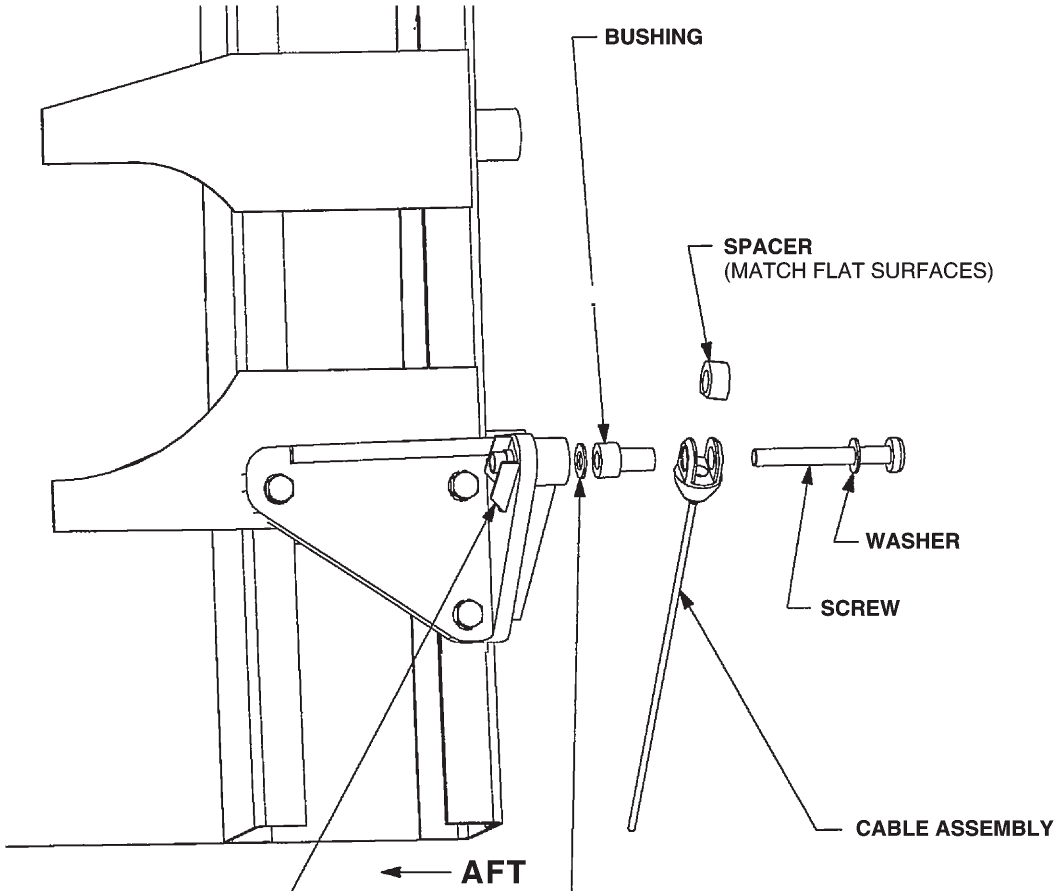


FORWARD CABLE UPPER MOUNTING



Effectivity
4636001 thru 4636445
with Kit No. 88423 Installed

Lower Cabin Door Support Installation
Figure 1 (Sheet 2 of 3)



AFT CABLE UPPER MOUNTING

Effectivity
4636001 thru 4636445
with Kit No. 88423 Installed

Lower Cabin Door Support Installation
Figure 1 (Sheet 3 of 3)

(c) Installation

1) Forward Cable

a) Route cable assembly from the spring motor assembly forward through the pulleys.

NOTE: Temporary removal of pulleys is required for routing.

b) Position the spring motor assembly with the rivet heads facing up. Rotate the spring motor assembly clockwise to fully wind the cable onto the drum. (See Procedure A, “Figure 1”.)

c) Insert shaft into existing base plate. With spring motor assembly clear of shaft and cable slack removed, insert installation tool (supplied with kit) into the top of the spring motor assembly and use it to rotate the bearing of the spring reel. Rotate clockwise 3/4 turn. (See Procedure B, Figure 1.)

d) Seat the spring motor assembly onto shaft using the installation tool as a guide. Maintain spring preload, rotate installation tool up to 1/4 turn to match flat surfaces on the shaft.

e) Remove the installation tool and install the required screw and washer. Torque 20–25 in.-lbs.

f) Mate the upper cable end with the lower end of handle and capture with then clevis pin.

g) Holding the clevis pin in place, secure with tumbler pin, compression spring, and socket head cap screw from top of handle. Torque 20–25 in.lbs. (See “Figure 1”, Sheet 2.)

NOTE: Ensure tumbler pin is properly seated in clevis pin groove.

2) Aft Cable

a) Route the cable from the spring motor assembly - aft through the pulleys.

NOTE: Temporarily remove pulleys to accomplish this.

b) Position the spring motor assembly as shown in Procedure C, “Figure 1”. Rotate the spring motor assembly counterclockwise to fully wind the cable onto the drum.

c) Insert shaft into existing base plate. With spring motor assembly clear of shaft and cable slack removed, insert installation tool (supplied with kit) into the top of the spring motor assembly and use it to rotate the bearing of the spring reel. Rotate counterclockwise 3/4 turn. (See Procedure D, “Figure 1”.)

d) Seat the spring motor assembly onto shaft using the installation tool as a guide. Maintain spring preload, rotate installation tool up to 1/4 turn to match flat surfaces on the shaft.

e) Remove the installation tool and install the shaft to secure the spring motor assembly to the door structure. Torque 20-25 in.-lbs.

f) Slide the damper assembly aft onto the mounting shaft, while simultaneously rotating damper assembly to facilitate nesting of the inner race of the damper assembly with roller clutch component of spring motor assembly.

g) Place washer onto mounting shaft and secure the entire assembly with retaining ring.

h) Secure the upper cable end to the fuselage structure with the screw, washers, spacer, and bushing.

3) Reinstall the cable stop(s).

4) If removed, reinstall existing decal holder(s).

(d) Final Adjustment - Operational Check

When the lower entry door is correctly installed, the proper door opening sequence is as follows:

1) From inside a closed cabin, unlatch the upper door and with a gentle motion, push the upper door outward roughly 12 inches, allowing the gas spring to lift the upper door. Note that the opening time is approximately three (3) seconds.

2) Unlatch the lower door, and using the same motion that was used to open the upper door, push the lower door outward roughly 12 inches. The rotary viscous damper will engage to slow the free fall of the lower door, adjust the number of turns on the spring motor assemblies as needed to achieve an opening time of approximately three (3) seconds, use installation tool as described in Installation, above.

3) To close the lower door from inside the cabin, face the exit, grasp the handle with the right hand and lift the door to within reach of the left hand, approximately 6 inches from the fully closed and latched position, holding the door with the left hand, ease the handle back into its stowed position in the eye bolt (with the handle oriented vertically), pull the door the remaining distance inward and latch securely.

4) Unlatch and open the lower door. When the lower door is in the down position, readjust the cable stops. The swaged cable ball ends on the cable assembly must make contact with the cable stops. Adjust position of cable stops to ensure simultaneous contact at forward and aft cable locations.

(e) Reinstall all interior panels. Trim plastic interior panels as required to allow the eye bolt and bushing to achieve metal-to-metal contact with the existing airframe-mounted brackets.

C. Lower Door Half (PA-46R-350T)

When open, the lower door half is supported by a cable assembly attached to lugs on the door's frame.

(1) Set-up

Open both upper and lower door halves. Support the lower door half to relieve the weight of the door from the hinge and the cable assemblies.

(2) Removal

(a) Complete set-up, above.

(b) Remove the cable retaining bolts, washers, and nuts from the door lug and door frame lug.

(c) Remove the cable assembly.

(3) Installation

(a) Attach the eye end of the cable to the door frame lug.

(b) Attach the yoke end of the cable to the door lug.

4. Door Latching Mechanism

A. Description

The upper door locking mechanism consists of external and internal handles, which actuate two locking pins in the lower corners of the door and two locking hooks, which protrude out of the lower edge of the door. When the handle is rotated to the open position, the guide pins and latch hooks are disengaged and withdrawn into the door.

An auxiliary lock is installed in the upper door, under the interior handle, to prevent inadvertent opening of the door. When opening the door from inside the airplane, the auxiliary lock handle must be pulled inboard, to allow the door handle to rotate.

The lower door locking mechanism is similar to the upper mechanism, except that it has no latch hooks and only one handle. Lifting the latch handle up withdraws the two pins into the door.

B. Upper Door

See “Figure 2”.

(1) Removal

(a) To remove the door handle bellcrank:

1) Remove the pin which retains the interior door handle to the bellcrank assembly. Remove the handle.

2) Remove the bellcrank support from the interior of the door by removing the rivets from the bellcrank support.

3) Remove the pins which attach the bellcrank to the pushrods.

4) Remove the bellcrank retaining screw from the bellcrank.

5) Pull the bellcrank assembly from the door.

(b) To remove the pushrod assemblies and the door retaining pins:

1) Remove the pin which connects the pushrod assembly to the bellcrank assembly.

2) Disconnect the pins and washer which secures the pushrod assembly to the door retaining pins.

3) Slide the pushrod and door retaining pin out of the door.

4) Repeat Steps 1, 2, and 3 for the remaining side.

(2) Installation

(a) To install the door handle bellcrank:

1) Position the bellcrank assembly in the door. Ensure that a washer is at each end of the bellcrank.

2) Secure the bellcrank assembly with the screw, nut and washer.

3) Attach the pushrod clevises to the bellcrank with the pins and washer.

4) Rivet the interior bellcrank support to the door.

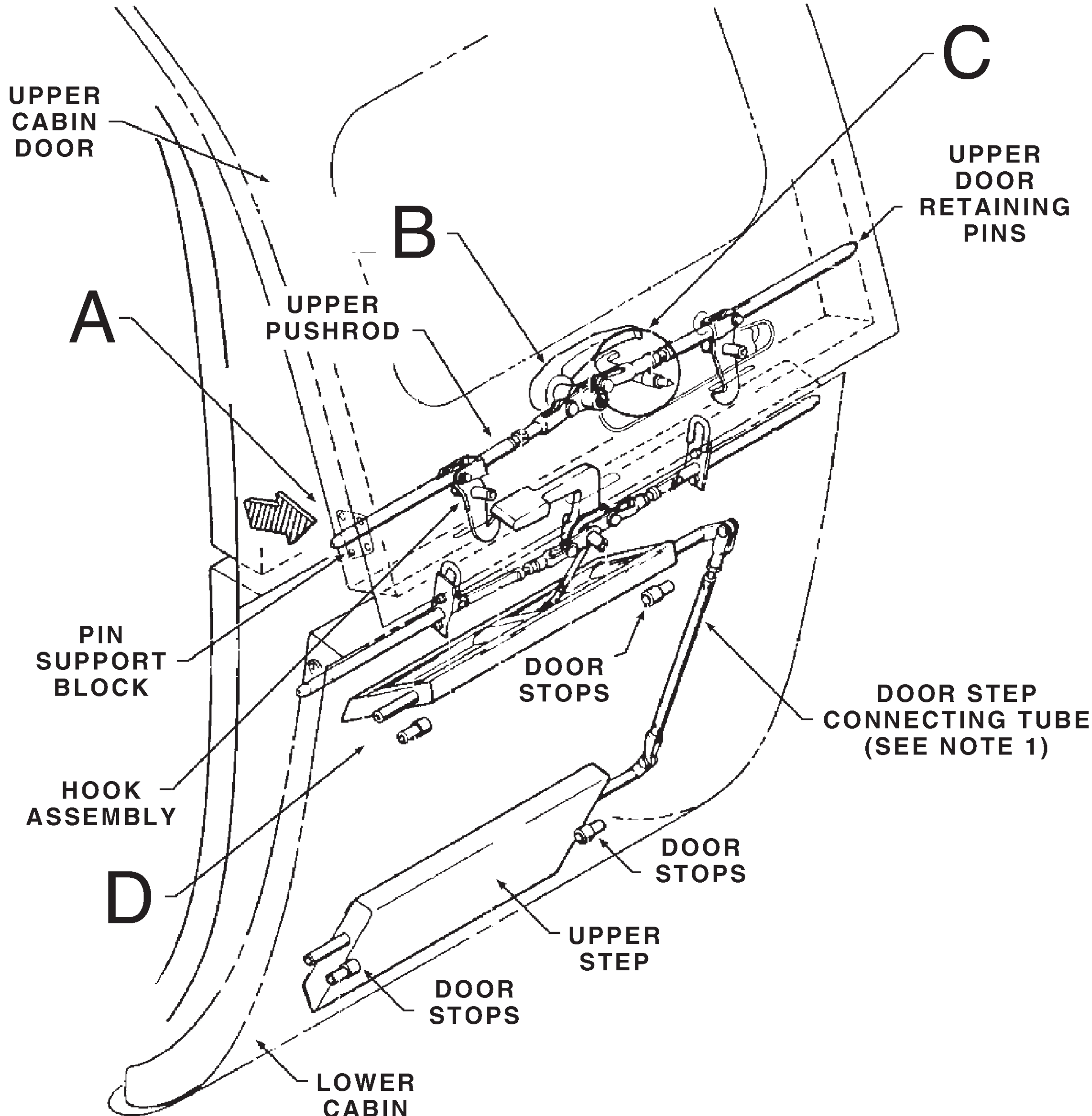
5) Attach the door handle to the bellcrank assembly with the pin.

(b) To install the door retaining pins and pushrod assemblies:

1) Slide the door retaining pin into position.

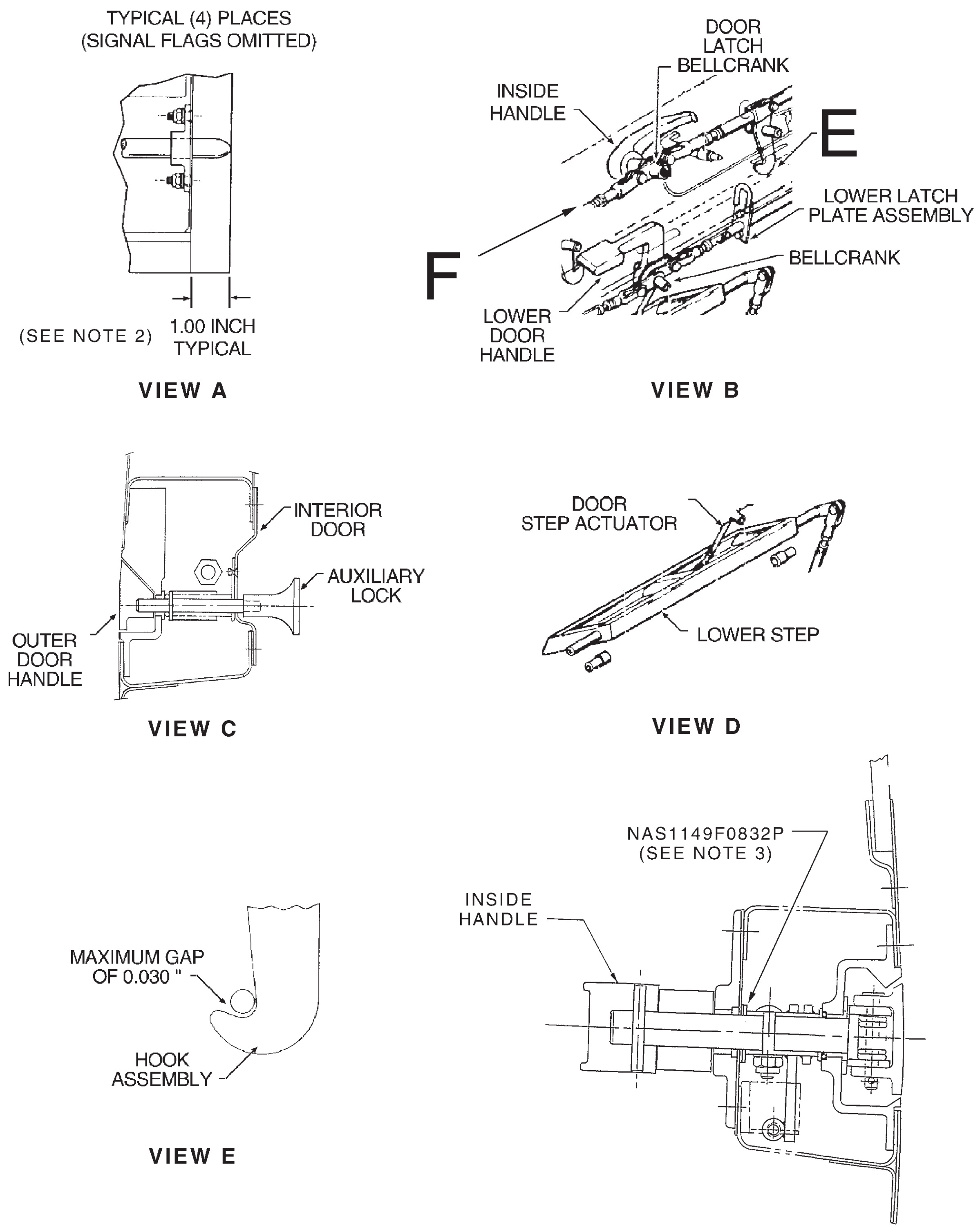
2) Place the pushrod assembly into position (the threaded section should point towards the door handle).

3) Align the holes in the pushrod clevis, door retaining pin and latch hook. Secure with the pins and washer.



NOTES:
1. With door handle in the open position, adjust steps to open up against stops.
2. Adjust upper and lower pushrods so that door retaining pins protrude 1.00 inch from door frame. (View A)
3. Install washers as required up to four (4) maximum to fill gap and minimize end play on shaft. Ensure free rotation of shaft and proper operation of latch mechanism.

Door Locking Mechanism
Figure 2 (Sheet 1 of 2)



Door Locking Mechanism
Figure 2 (Sheet 2 of 2)

4) Attach the remaining pushrod clevis end to the door handle bellcrank with the pins and washers.

NOTE: The door retaining pins should protrude out of the door 1.00 inch with the door handle in the closed position. Make adjustments at the pushrod assembly as required.

(3) Adjustment

(a) Shut and latch both doors.

(b) Check gap of hook and latch plate.

(c) Hook may be filed .053” max. (maintaining the same ramp contour) to provide engagement with plate on lower door assembly. A gap of up to 0.030” maximum between hook and latch plate (with hook fully engaged and without cabin pressure) is acceptable.

C. Lower Door

See “Figure 2”.

(1) Removal

(a) To remove the door handle bellcrank:

1) Disconnect the pushrod clevises from the bellcrank by removing the pins and washer.

2) Remove the bellcrank retaining bolt.

3) Remove the rivets which retain the interior bellcrank support to the door channel.

4) Remove the screws and nuts which retain the handle to the bellcrank.

5) Remove the bellcrank.

(b) To remove the pushrod assemblies and the door retaining pins:

1) Disconnect the pushrod clevis from the handle bellcrank by removing the pins and washer.

2) Disconnect the pushrod clevis from the door retaining pin by removing the pins and washer.

3) Slide the retaining pin out of the door and remove the pushrod assembly.

(2) Installation

(a) To install the door handle bellcrank:

1) Position the bellcrank in door.

2) Attach the lever to the bellcrank with the screws, washers and nuts.

3) Ensure that washers are positioned at each end of the bellcrank. Insert the bolt through the interior bellcrank support.

4) Rivet the support in place and tighten the bolt.

(b) To install the pushrod assemblies and the door retaining pins:

1) Position the pushrod assembly in the door. The threaded portion of the pushrod should point towards the bellcrank assembly.

2) Secure the pushrod with the pins and washer.

3) Slide the door retaining pin into the door.

4) Attach the door retaining pin to the pushrod with the pins and washer.

NOTE: The door retaining pins should protrude from the edge of the door approximately 1.00 inch. Make adjustments to the pushrod assembly as required. After installation mechanism must rotate and operate easily through full cycle without friction and binding.

5. Door Seal

NOTE: In S/N's 4636001 thru 4636313 only, if door seal has been damaged by upper door latch pins being extended while the door is moved towards the closed position, install Piper Kit No. 767-387 to preclude reoccurrence.

A. Removal

To remove the door seal, pull the cemented surfaces free of the retainer and/or door edge, applying MEK as needed to help loosen the bond.

B. Installation

Refer to "Figure 3".

NOTE: Piper Kit No. 88440-002 includes door seal P/N 82780-002 as well as additional parts required to properly install the door seal.

(1) Remove any traces of the previous seal and cement.

(2) Thoroughly clean the areas to be cemented with MEK.

(3) Bond the seal to the door and/or angle per Figure 3 and the following.

C. Bonding

(PIR PPS45009, Rev. E.)

(1) Use Adhesive Silicone RTV 157 (Piper Code 279-021).

(2) Clean surfaces to be bonded or sealed with a clean, lint-free cloth or wiper dampened with isopropyl alcohol. Wipe dry using a dry wiper before the alcohol evaporates.

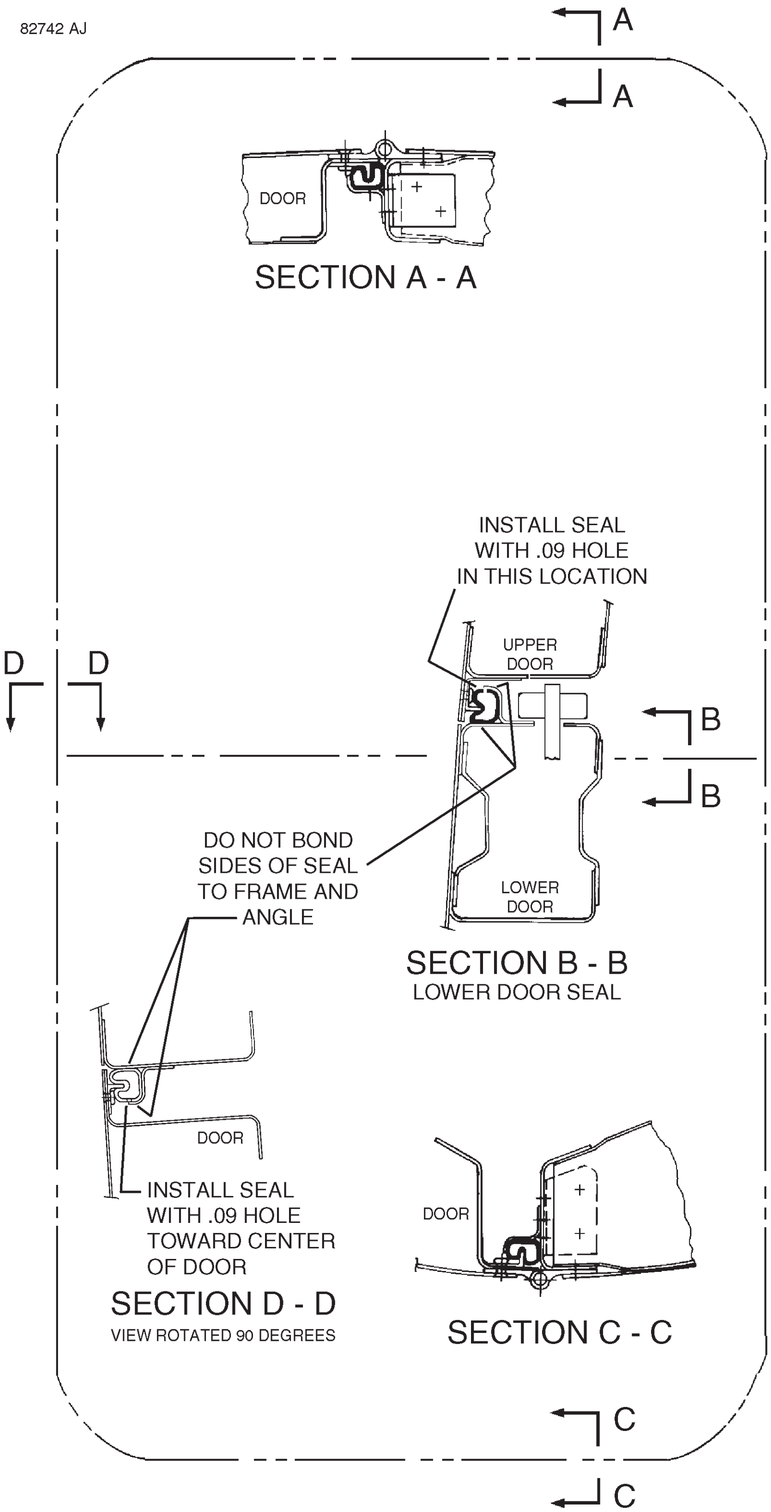
(3) Apply adhesive to only one of the surfaces to be joined. Thickness of the adhesive shall be 0.015 inch minimum and 1/8 inch maximum. Press surfaces to be joined together firmly. No "squeeze out" of adhesive should be left around the edges of the joint.

(4) Apply sealant as required to specified areas, however the maximum section of the sealant shall be 1/8 inch.

D. Cure

Handle bonded and sealed assemblies with care. Adhesion will occur in 24 hours, and full cure will occur in seven days at 75° F and 50% relative humidity.

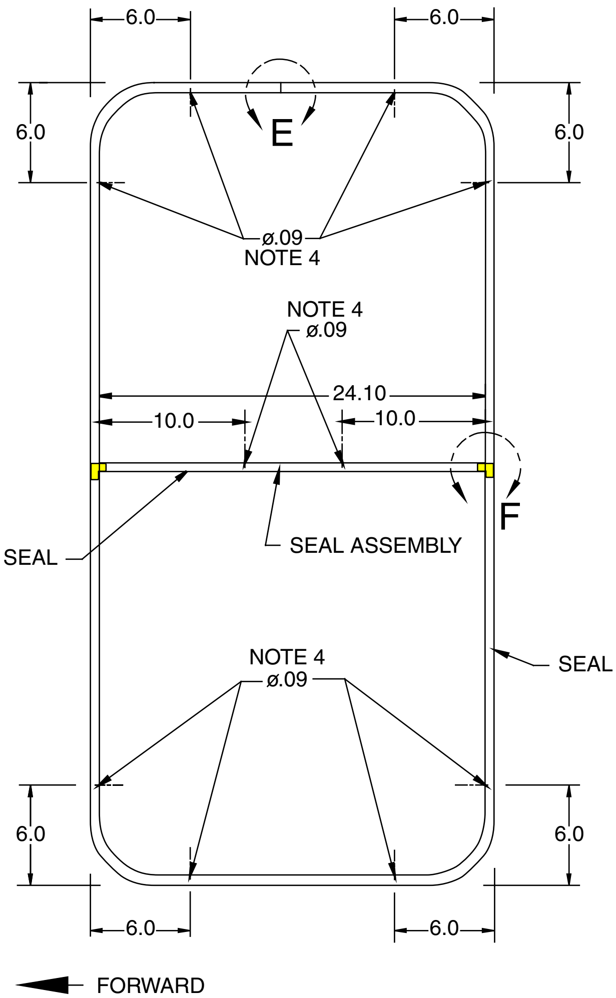
NOTE: Higher temperatures and relative humidity will shorten this cure time.



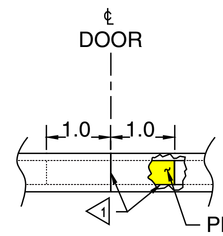
NOTE: Bond seal to angle with RTV157 after cleaning surface with isopropyl alcohol to remove dirt, oil, and/or grease.

Door Seal Installation
Figure 3 (Sheet 1 of 2)

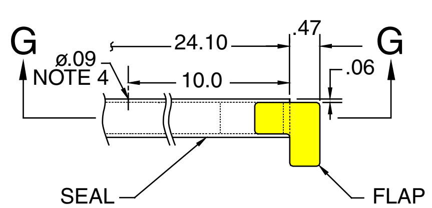
82780 R



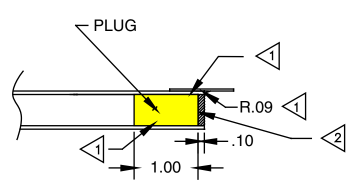
VIEW LOOKING INBOARD
CABIN DOOR
SEALS SHOWN AS INSTALLED



VIEW E
BONDING OF SEAL ENDS TOGETHER



VIEW F
TYPICAL BOTH ENDS
LOWER CABIN DOOR



SECTION G - G
TYPICAL BOTH ENDS
LOWER CABIN DOOR

Door Seal Installation
Figure 3 (Sheet 2 of 2)

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EMERGENCY EXIT

1. Emergency Exit Door

The airplane's emergency exit door is located on the right-hand side of the airplane, at the center seat.

A. Removal

To remove the emergency exit door, remove clear cover over Emergency Door Latch Handle, pull down on the release at the top of the door, bring the top of the door slightly down and lift the entire door towards the center of the aircraft.

B. Installation

To install the door, insert the lower lip of the door on the outboard side of the two stops, center door to achieve equal clearance fore and aft, and push top of the door outwards until the handle snaps into place on the catch assembly.

C. Functional Test

(PIR PPS60233, Rev. A.)

From inside the aircraft, measure the clearance between the liner panel attached to the Emergency Door, and the surrounding liner panel that is attached to the airframe, at the upper 4-inch section of this interface, as identified in Figure 1 and View A-A. If less than .10 inches (one-tenth inch) exists, rework liner panel attached to Emergency Door, according to specifications.

NOTE: If the Emergency Door contains an interior light, it must be disconnected prior to panel rework and functional test, in order to avoid damage to wiring. Locate pin and socket connector, which is mounted on the wire bundle, 7 inches from the bulb. Separate connector as needed, being sure to re-connect upon completion of functional testing.

(1) Removal

Remove door as noted above. Prior to door removal, use masking tape on the exterior of the door to establish temporary witness marks, which will be used later for alignment during reinstallation.

(2) Installation

Install door as noted above. For ease of reinstallation, door seal may be lubricated sparingly with silicone grease such as Parker Super O-Lube (Piper p/n 197-507) or equivalent.

(3) Leak Test

Following re-installation of Emergency Exit Door, verify water-tightness (See Leak Test in 56-00-00). Temporarily remove interior components as needed to examine for evidence of water intrusion.

2. Seal

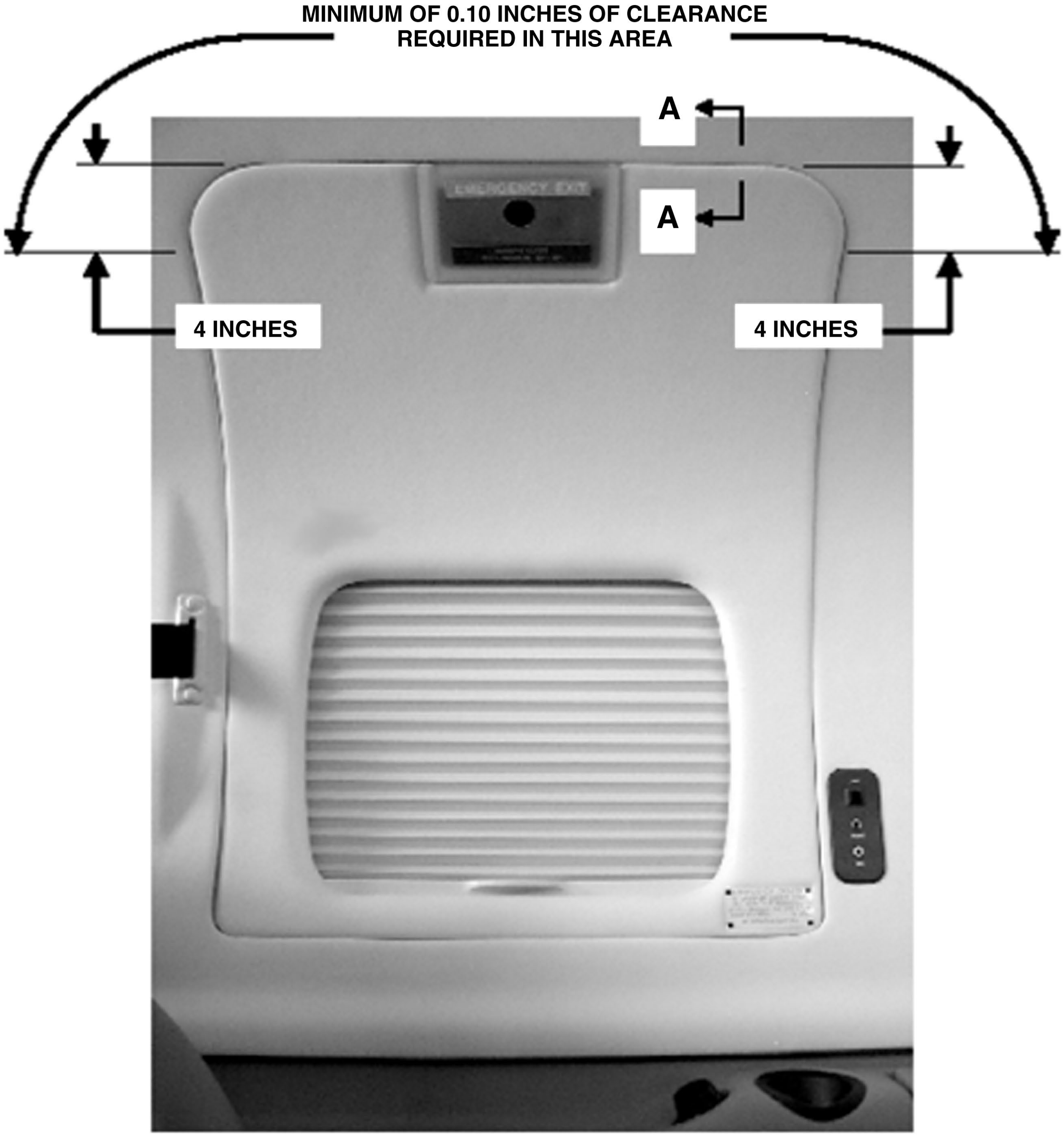
A. Removal

Remove the seal by pulling it away from the skin.

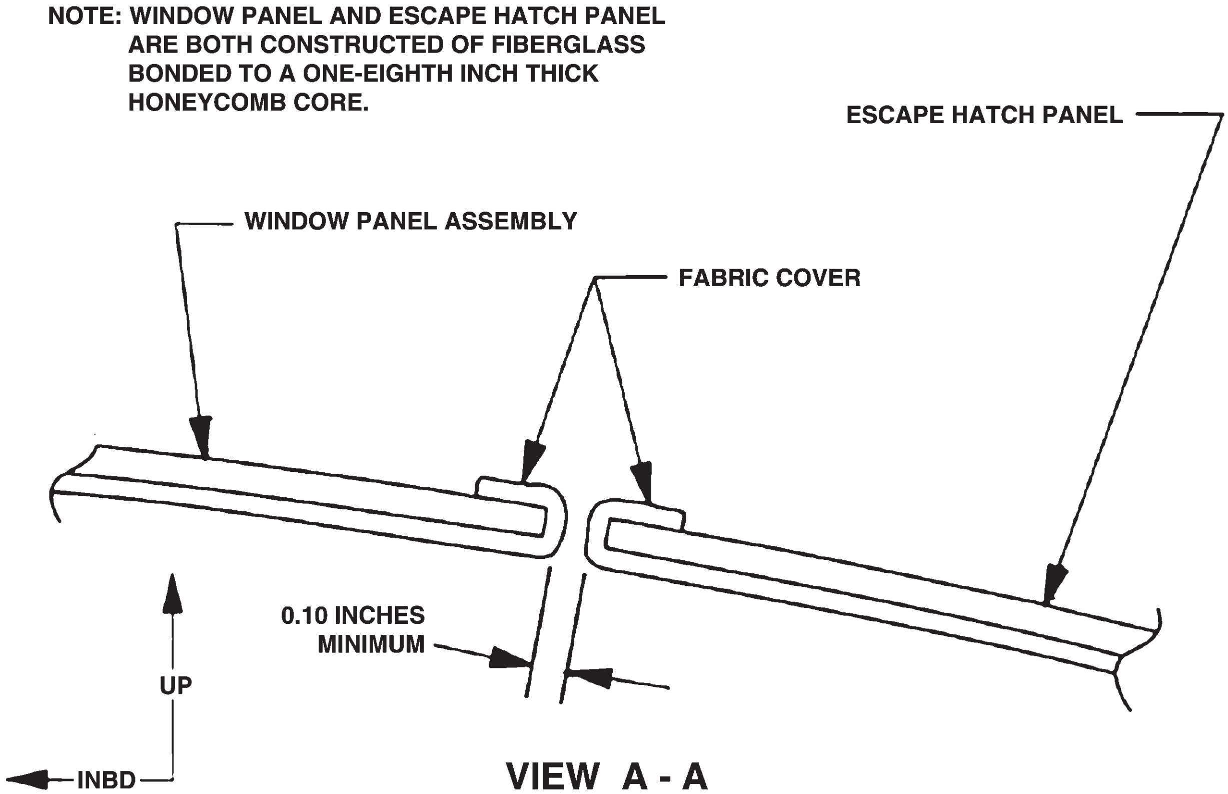
B. Installation

(1) Prepare the surface by removing any remaining trace of the old seal or old cement, using a clean rag moistened with MEK.

(2) Apply a small bead of RTV157 Grey (P/N 279-021) adhesive or equivalent on the outside radius of the door seal flange and cement the seal in place.



Emergency Exit Door Trim
Figure 1 (Sheet 1 of 2)



Emergency Exit Door Trim
Figure 1 (Sheet 2 of 2)

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CARGO

Baggage Compartment Door

The forward baggage compartment is accessed via a separate door located on the left-hand side of the fuselage, between the forward pressure bulkhead and the firewall.

A. Removal

(1) Unlock and unlatch the door.

(2) Disconnect the door support from the door.

(3) Lift the door slightly and support it while removing the hinge pin.

B. Installation

(1) Align the door hinge with the hinge on the fuselage and install the pin. It is recommended that a new pin be used.

(2) Reconnect the door support to the door.

(3) Lubricate the pin and close the door.

C. Seal Installation

(PIR PPS45009, Rev. E.)

NOTE: Piper Kit No. 88441-002 includes baggage door seal P/N 82780-004 as well as a plug and instructions required to properly install the door seal.

(1) Remove any traces of the previous seal and cement.

(2) Clean surfaces to be bonded or sealed with a clean, lint-free cloth or wiper dampened with isopropyl alcohol. Wipe dry using a dry wiper before the alcohol evaporates.

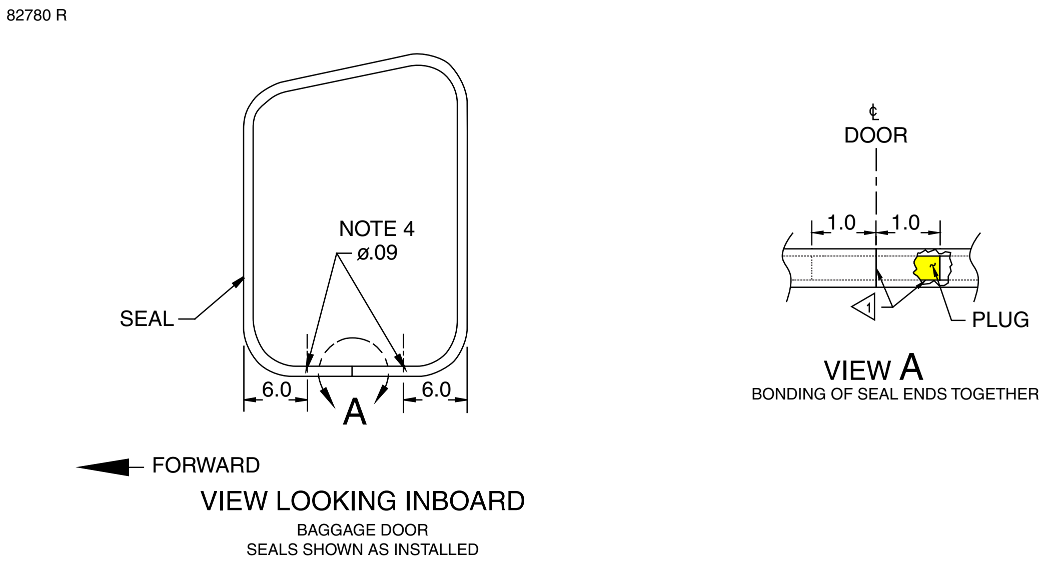
(3) Bond the inboard surface of the seal to the door and/or angle per “Figure 1” and steps (4) thru (7) below.

(4) Use Adhesive Silicone RTV 157 (Piper Code 279-021).

(5) Apply adhesive to only one of the surfaces to be joined. Thickness of the adhesive shall be 0.015 inch minimum and 1/8 inch maximum. Press surfaces to be joined together firmly. No “squeeze out” of adhesive should be left around the edges of the joint.

(6) Apply sealant as required to specified areas, however the maximum thickness of the sealant shall be 1/8 inch.

(7) Handle bonded and sealed assemblies with care. Adhesion will occur in 24 hours, and full cure will occur in seven days at 75°F and 50% relative humidity. Note higher temperatures and relative humidity will shorten this cure time.



Baggage Door Seal Installation
Figure 1

MONITORING AND OPERATION

1. Cabin Door Ajar Warning

   A. Description

   The cabin door ajar warning system consists of two switches mounted in the W.L. 76.15 and W.L. 71.47 longerons. These switches are activated by the two forward door locking pins pushing on two switch actuating pins. A DOOR AJAR light is provided in the annunciator panel.

   NOTE: On Garmin G1000/G1000 NXi equipped aircraft, the "door ajar" CAS message is displayed on the MFD.

   B. Troubleshooting

   See "Chart 1".

CHART 1
TROUBLESHOOTING CABIN DOOR AJAR WARNING SYSTEM

Trouble	Cause	Remedy
Cabin door unsafe light (DOOR AJAR) will not go out.	Latching relay sticking in the open position.	Lubricate the mechanical parts (sparingly).
	Switch or switches in door frame out of adjustment.	Adjust.
	Short to ground.	Check wiring and repair.
Cabin door unsafe light (DOOR AJAR) will not come on.	Circuit breaker open.	Reset circuit breaker.
	Broken wire or loose connection.	Check wiring and repair.

   C. Switches

   (1) Removal

   (a) Gain access to the switches. They are located approximately four (4) inches in front of each of the forward door locking pins.

   (b) Disconnect the two electrical leads (per switch).

   (c) Loosen the jam nut on the switch and turn the switch body out of the pin support tube assembly.

   (2) Installation

   (a) Turn the switch body into the pin support tube assembly. Do not tighten the jam nut at this time.

   (b) Adjust the switches per the following paragraph.

   (c) Reconnect the two electrical leads (per switch).

(3) Adjustment

(a) Ensure that the door retaining pins protrude out the side of the door 1.00 inch when the latch mechanism is in the closed position. (See View A, “Figure 1”, 52-10-00.)

(b) Close the door and turn the latch mechanism to its locked position.

(c) Turn the switch body into the pin support tube assembly until the switch activates.

(d) Tighten the switch jam nut.

(e) Connect the two leads to the switch.

(f) Repeat Steps (a) thru (e) for the other switch.

(4) Checks

Periodically check the action of the cabin door switches as follows:

(a) Insert a small blunt object into the pin support tube receptacle.

(b) Push on the switch actuating pin until an audible click is heard. Release the pin, noting another audible click.

(c) Replace any switch in which the click action and mechanical function is doubtful.

(d) Ensure that the DOOR AJAR light in the annunciator panel comes on when the switch is activated.

2. Baggage Compartment Door Ajar Warning

The baggage door ajar switch is located behind the baggage door latch in the longeron at W.L. 55.46. A DOOR AJAR light is provided in the annunciator panel (conventional instrument or Avidyne Entegra equipped aircraft) or a “DOOR AJAR” CAS message is displayed (Garmin G1000/G1000 NXi equipped aircraft).

A. Adjustment

Adjust the switch by loosening one of the two locking nuts on the switch body and turning the switch in or out until a slight lifting of the baggage door latch activates the switch.

B. Checks

Periodically check the action of the baggage door switch as follows:

(1) Slightly lift the baggage door latch, listening for audible clicks when the switch activates and deactivates.

(2) Ensure that the DOOR AJAR light in the annunciator panel (or a “DOOR AJAR” CAS message) comes on when the switch is activated.

CHAPTER

53

FUSELAGE

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CHAPTER 53

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
53-List of Effective Pages	1	Jul 1/10			
	2	Jul 1/10			
53-Table of Contents	1	Jul 1/10			
	2	Jul 1/10			
53-20-00	1	Jul 1/10			
	2	Jul 1/10			

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CHAPTER 53 - FUSELAGE

TABLE OF CONTENTS

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MAIN

Drain Hole Inspection

A. In S/N's 4692001 thru 4692135 only, for airplanes which have not yet complied with Part II of Piper Service Bulletin No. 1211:

(1) Prior to each flight;

(2) Immediately following flight into rain;

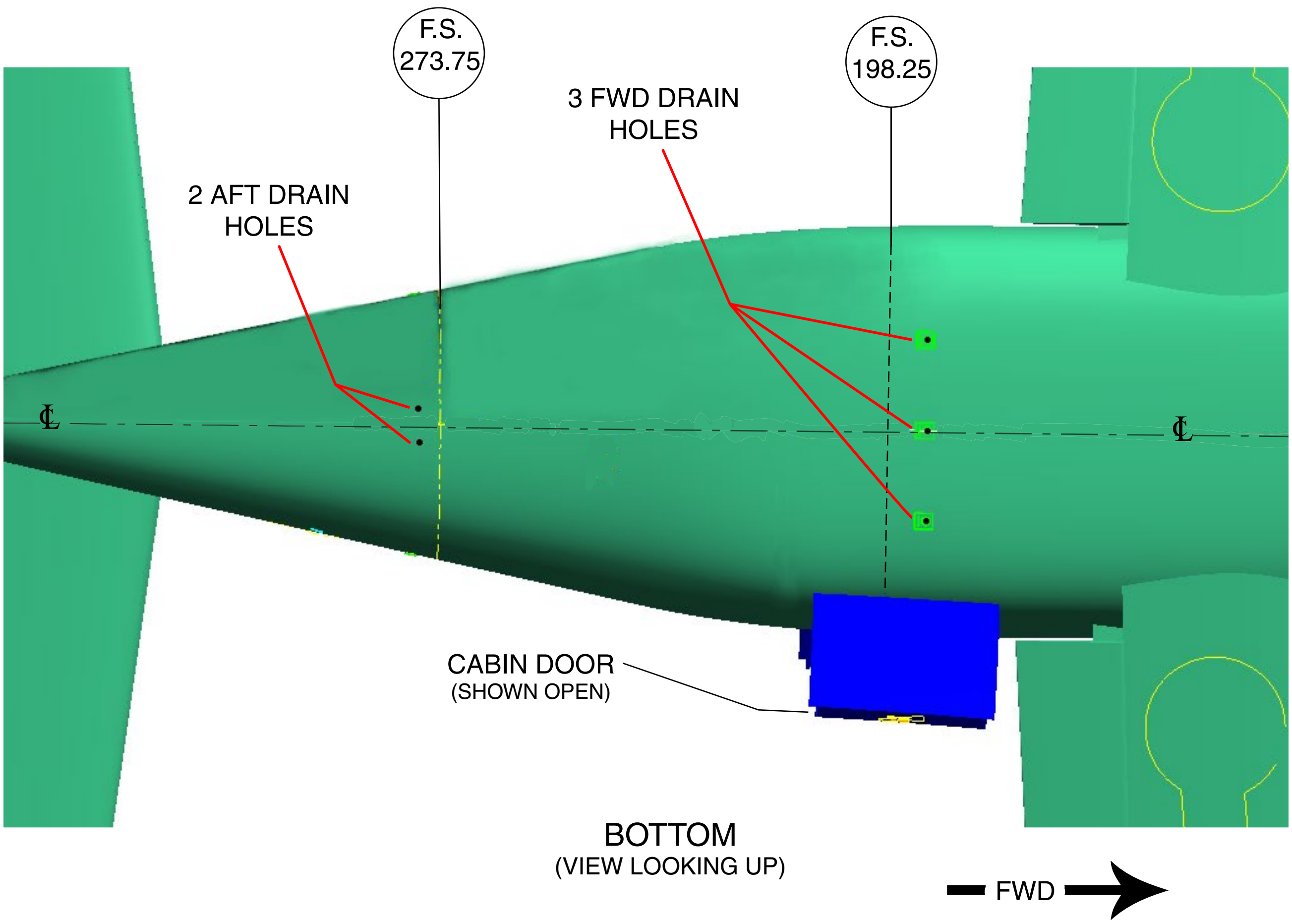
(3) Immediately following airplane wash;

(4) For aircraft stored uncovered, within one week after each exposure to rainy environmental conditions;

Carefully insert a blunt, rounded-end tool into all five drain holes shown in "Figure 1". Ensure that they are not blocked by any foreign matter and are free to drain.

B. In all airplanes, each 100 hours or annually, whichever comes first:

Carefully insert a blunt, rounded-end tool into all five drain holes shown in "Figure 1". Ensure that they are not blocked by any foreign matter and are free to drain.



Drain Hole Inspection
Figure 1

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CHAPTER

55

STABILIZERS

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CHAPTER 55

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
55-List of	1	Nov 30/17			
Effective Pages	2	Sep 15/09			
55-Table of Contents	1	Nov 30/17			
	2	Sep 15/09			
55-00-00	1	Nov 30/17			
	2	Nov 30/17			
	3	Sep 15/09			
	4	Sep 15/09			
55-10-00	1	Nov 30/17			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Sep 15/09			
55-20-00	1	Feb 15/14			
	2	May 31/11			
	3	Sep 15/09			
	4	Nov 30/17			
	5	Nov 30/17			
	6	Nov 30/17			
	7	Nov 30/17			
	8	Nov 30/17			
55-30-00	1	Nov 30/17			
	2	Sep 15/09			
55-40-00	1	Nov 30/17			
	2	May 31/11			

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CHAPTER 55 - STABILIZERS

TABLE OF CONTENTS

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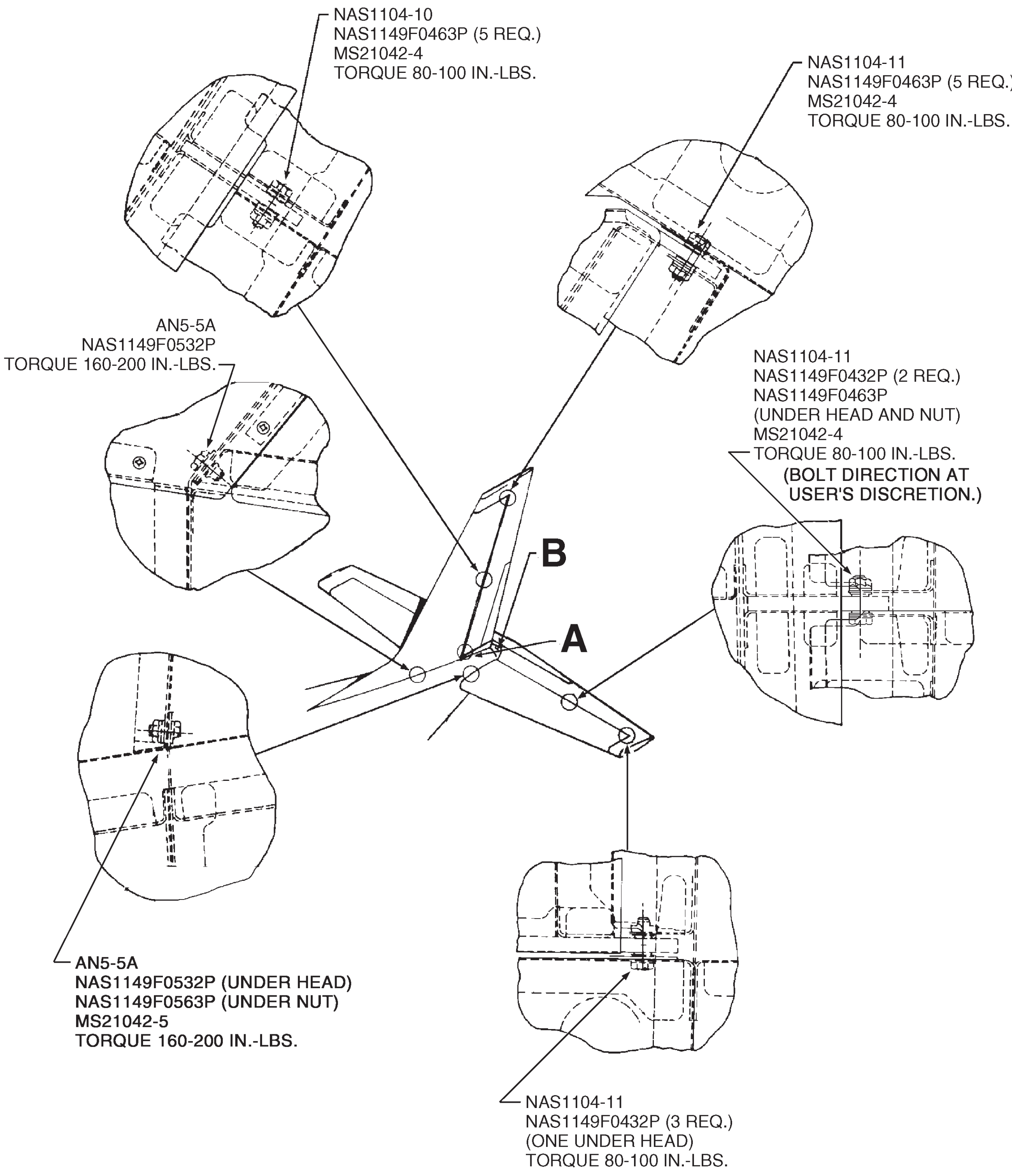
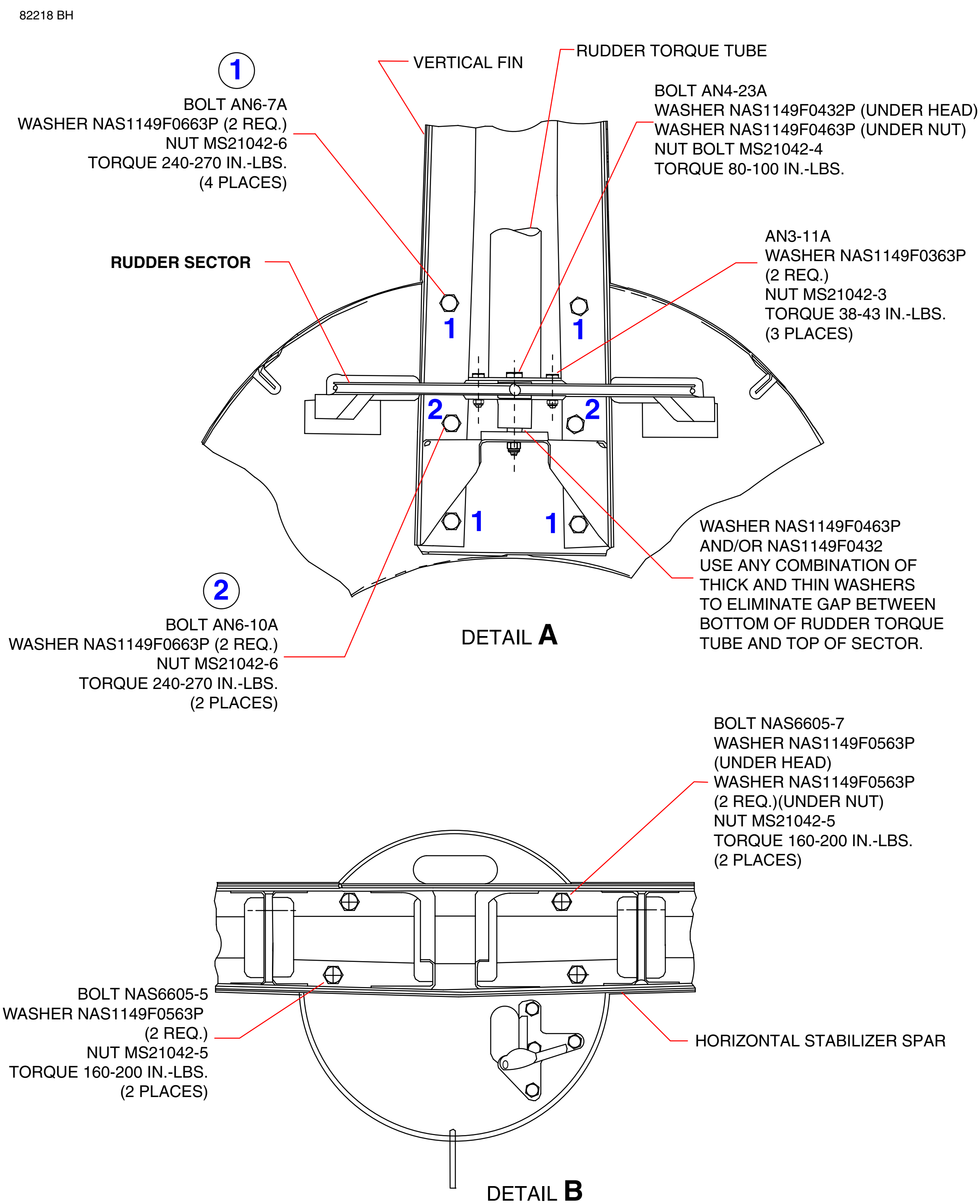
GENERAL

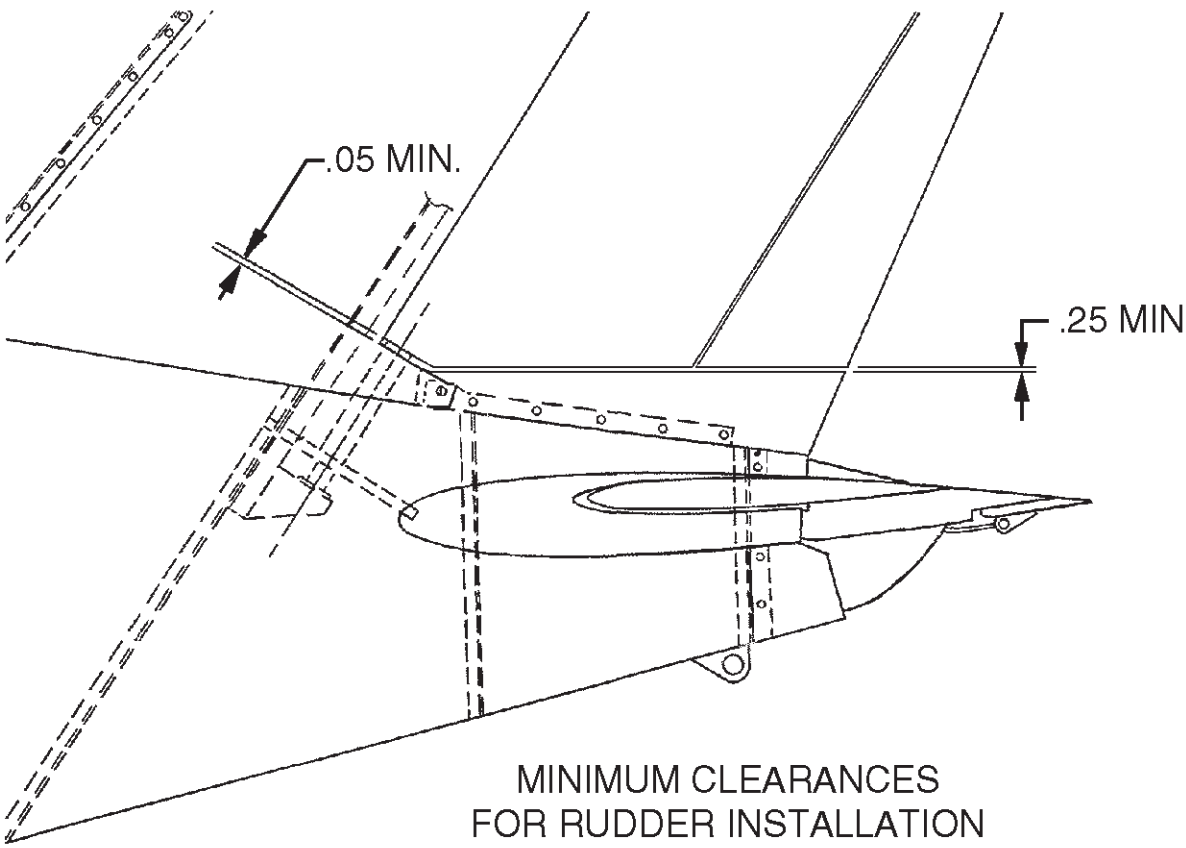


NOTE: A & B ON SHEET 2.

Empennage Installation
Figure 1 (Sheet 1 of 3)



Empennage Installation
Figure 1 (Sheet 2 of 3)



Empennage Installation
Figure 1 (Sheet 3 of 3)

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HORIZONTAL STABILIZER

1. Horizontal Stabilizer

Refer to “Figure 1”, 55-00-00.

A. Removal

(1) Remove the elevator (refer to 55-20-00).

(2) Remove the nut, bolt and washer that attach the stabilizer forward support to the bulkhead (each side).

(3) Support the stabilizer and remove the nuts, bolts and washers that attach the stabilizer spar to the bulkhead. Remove the stabilizer.

B. Installation

(1) Position the horizontal stabilizer on the tail section.

(2) Attach the horizontal stabilizer spar to the bulkhead. Torque the four nuts as specified in “Figure 1”, 55-00-00.

(3) Attach the two horizontal stabilizer forward supports to the bulkhead. The bolts should be installed with their heads pointing aft. Torque the bolts as specified in “Figure 1”, 55-00-00.

(4) Install the elevator (refer to 55-20-00).

(5) Check elevator trim and elevator operation (refer to 27-30-00).

2. Attach Fittings Corrosion Control

Each five years, inspect the attach fittings shown in “Figure 1” on page 55102 and “Figure 2” on page 55103 for corrosion and condition. Repair/replace as required. When inspection and repair/replacement is completed, reapply Dinitrol AV 8 (P/N 89500-800) Corrosion Inhibiting Compound.

NOTE: For all fittings, inspection of the exposed portion is sufficient unless obvious or extensive corrosion is apparent.

NOTE: Any part exhibiting flaking of the metal due to rust must be replaced.

NOTE; If aluminum skin or structure is corroded (beyond a light surface oxidation), the area must be repaired or replaced.

NOTE: On any part being replaced, dip rivets in waterborne, chromated, fluid resistant epoxy primer (i.e. - PRC Desoto) or equivalent primer and install wet.

A. When inspection and repair or replacement are completed, allow sufficient time for primer to dry, and apply Dinitrol AV 8 to:

(1) Forward horizontal stabilizer attach points. Inside and outside fuselage. See “Figure 2” on page 55103 to add an access hole.

(2) Aft horizontal stabilizer attach points.

(3) Entire surface of tail cone interior.

CAUTION: READ PRECAUTIONS ON CONTAINER LABEL AND WEAR A CHARCOAL FILTER MASK WHEN APPLYING DINITROL AV 8.

B. Apply Dinitrol by sprayer or brush. Use an air or airless spray gun, if available. A hand aspirator bottle may be used if a sufficient fan spray pattern can be developed. Product should not be thinned.

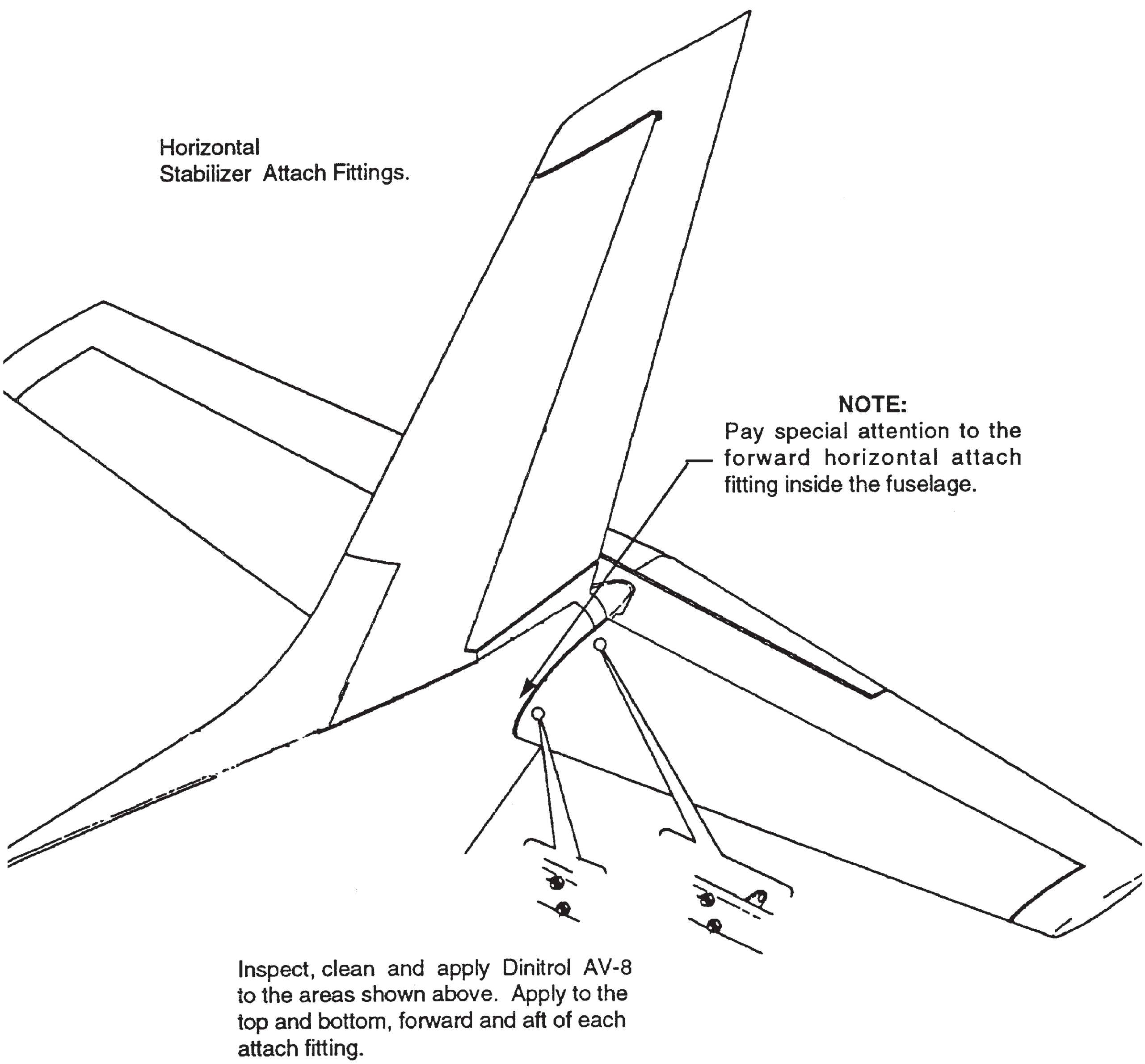
C. Cover safety and outflow valves, air conditioning condenser, and other components on the pressure bulkhead. Cover autopilot servos, electrical connectors, and cable pulleys.

WARNING: DO NOT GET DINITROL INTO AUTOPILOT SERVOS OR CLUTCHES.

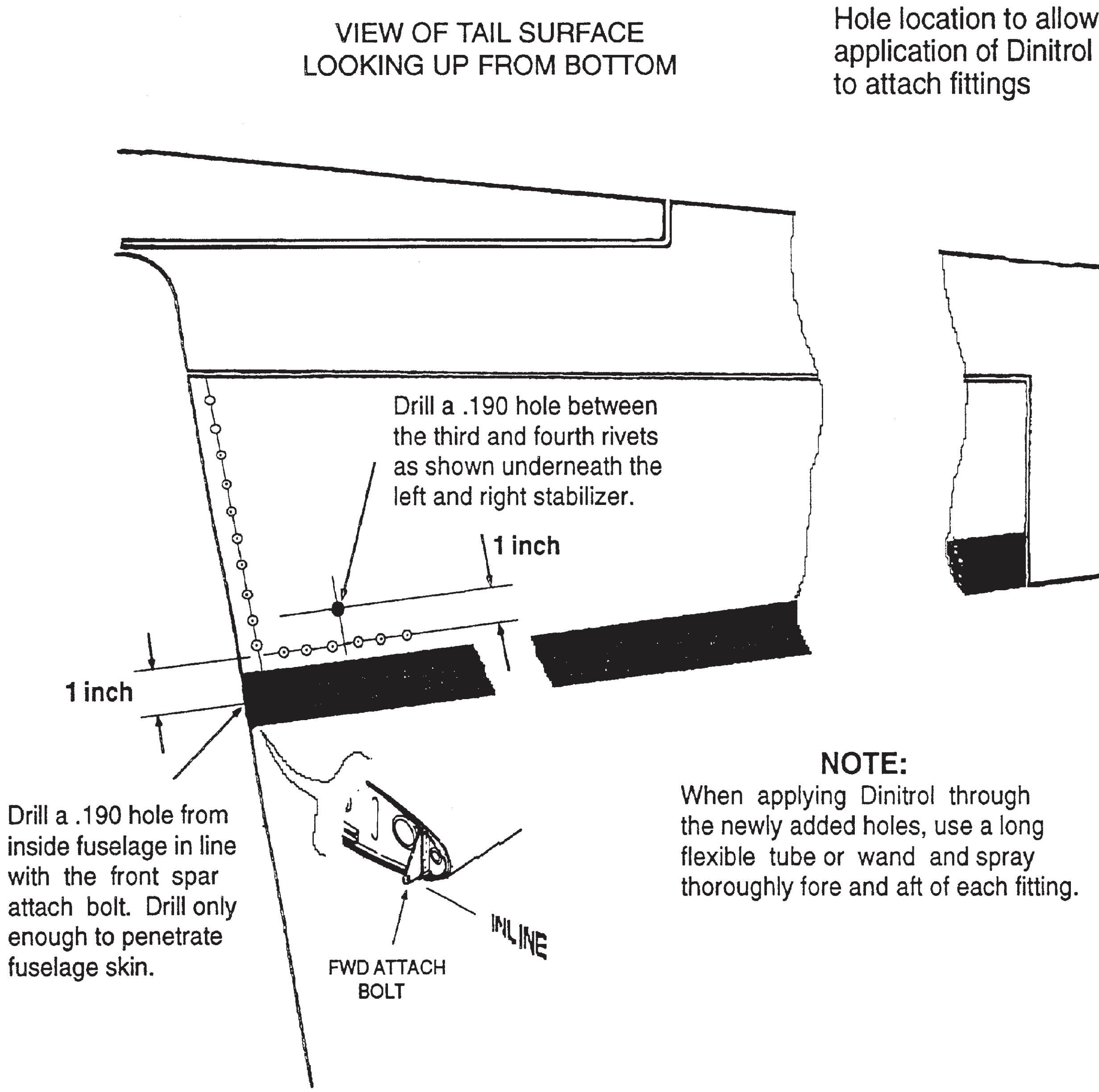
D. Spray directly into faying surfaces and mating points.

E. Ensure entire attach fitting is treated. Spray inside where the fitting is attached to the structure.

F. Apply to approximately .8 mils in thickness. Avoid puddling and running. Allow to dry 1 to 1 1/2 hours before airplane use. Full drying takes 6 to 8 hours.



Horizontal Stabilizer Attach Fittings
Figure 1



Horizontal Stabilizer Forward Attach Fitting Access
Figure 2

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ELEVATOR

1. Elevator

See “Figure 1”, 55-00-00.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

A. Removal

(1) Disconnect the elevator trim actuating rods at the clevis on the actuator.

(2) Disconnect the elevator actuating arm at the elevator actuating plate.

(3) Support the elevator and remove the attaching bolts.

(4) Remove the elevator from the airplane.

B. Installation

WARNING: IF THE ELEVATOR HAS BEEN REPLACED OR REPAINTED, OR THE TRIM TAB HAS BEEN REPLACED OR REPAINTED; THE ELEVATOR MUST BE BALANCED BEFORE INSTALLATION. SEE BALANCING, BELOW.

(1) Position the elevator in place on the horizontal stabilizer.

(2) Attach the elevator to the horizontal stabilizer. Torque the nuts as specified in “Figure 1”, 55-00-00.

(3) Connect the elevator actuating arm to the elevator actuating plate.

(4) Connect the elevator trim actuating rods to the actuator clevis.

(5) Check elevator trim and elevator operation (refer to 27-30-00).

C. Balancing

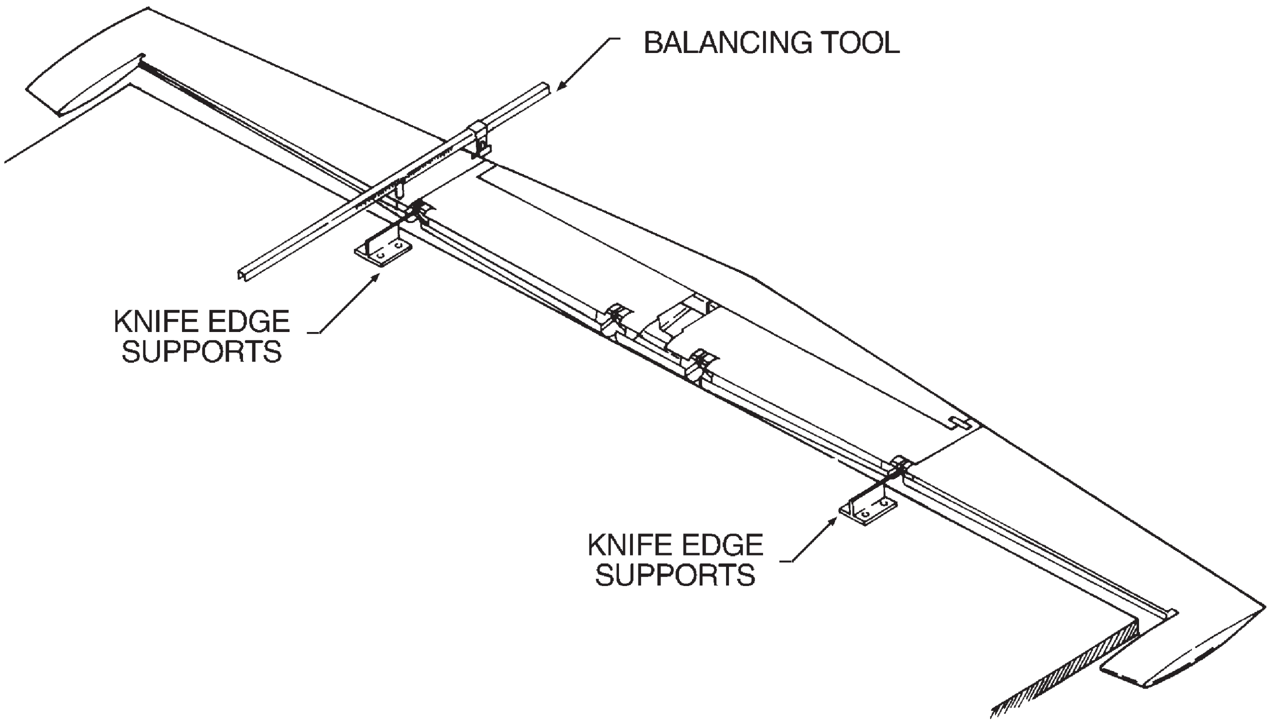
(PIR-PPS50060, Rev. K.)

See “Figure 1”.

To balance the elevator, the assembly must be complete and in its final flight configuration as specified under Balancing Equipment, below (without the tab pushrods, with elevator tips and all attaching screws). Before balancing, tape the trim tab in its neutral position, using a small piece of tape. Ensure that there are no drafts in the balancing room. Place the complete assembly on the knife edge supports of a balancer with the beam perpendicular to the hinge centerline. Do not place the tool on the trim tab. Calibrate the tool per Balancing Equipment, below. Read the scale when the bubble level has been centered by adjustment of the movable weight and determine the static balance limit (see “Chart 1”). There are no adjustment provisions on the elevator.

CHART 1
ELEVATOR STATIC BALANCE LIMITS

Leading Edge Heavy (Inch-Pounds)	Trailing Edge Heavy (Inch-Pounds)
—	-25 to -50
NOTE: Elevator is balanced with tab installed but without the tab pushrods.	



Balancing Elevator
Figure 1

2. Balancing Equipment

Balancing must be done using a suitable tool capable of measuring unbalance in inch-pounds from the centerline of the control surface hinge pin. See the tool configuration in “Figure 2”. Other tool configurations may be used if accuracy is maintained and recalibration capability is provided.

To use this tool:

A. Ensure that the control surface is in its final Flight configuration (i.e. - static wicks, trim tabs, and control surface tip (as applicable) should be installed). The surface should be painted, decals applied and trim/servo tabs should be in the neutral position.

NOTE: Because paint is a considerable balance factor, it is recommended that existing paint be removed prior to repainting a control surface.

B. Place hinge bolts through control surfaces and place control surface on a holding fixture.

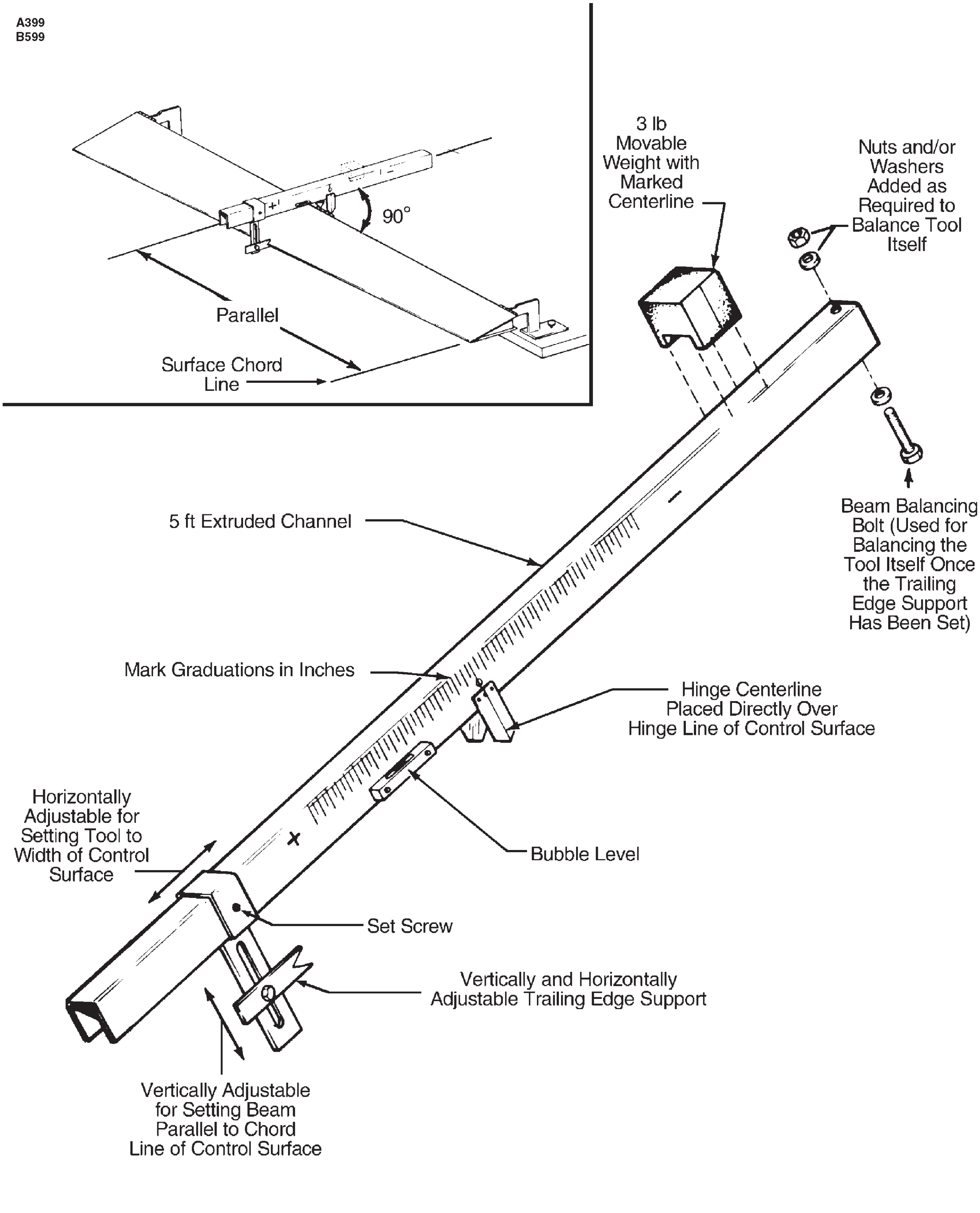
C. Calibrate the tool.

(1) Avoiding rivets, place the balancing tool on the control surface with the tool’s hinge centerline directly over the hinge line of the control surface.

(2) Adjust the movable trailing edge support to fit the width of the control surface. Tighten the set screw on the trailing edge support.

(3) Adjust the trailing edge support vertically until the beam is parallel with the control surface chord line.

(4) Remove the tool from the control surface and balance the tool itself by adding or removing nuts or washers from the beam balancing bolt. When balancing the tool, the movable weight must be at the bar’s hinge centerline.



Control Surfaces Balancing Tool
Figure 2

D. After balancing the tool, reattach it to the control surface per “Figure 1” and “Figure 2”. Keep the beam positioned 90° from the control surface hinge line.

E. Determine balance of control surface by sliding movable weight along the balance beam.

F. Read the scale when the bubble level has been centered. Multiply by three to determine inch-lbs. (i.e. - Since the movable weight weighs three pounds, every inch it is moved from the center of the beam equals three inch-lbs of force.)

3. Elevator Trim Tab Push Rods Attach Brackets

If the elevator trim tab push rods attach brackets are removed and replaced or reinstalled, ensure that a coat of waterborne, chromated, fluid resistant epoxy primer (i.e., PRC Desoto) is applied between the tab skin and the brackets to eliminate surface corrosion.

4. Elevator Trim Tab Rods Corrosion Control

Each five years, remove the elevator trim tab rods from the airplane and inspect as shown in “Figure 3” for corrosion and condition. Repair/replace as required. When inspection and repair/replacement is completed, reapply Dinitrol AV 8 (P/N 89500-800) Corrosion Inhibiting Compound.

NOTE: For all fittings, inspection of the exposed portion is sufficient unless obvious or extensive corrosion is apparent.

NOTE: Any part exhibiting flaking of the metal due to rust must be replaced. The elevator trim tab actuator push rods must be replaced if rust is evident extending from inside the tubes.

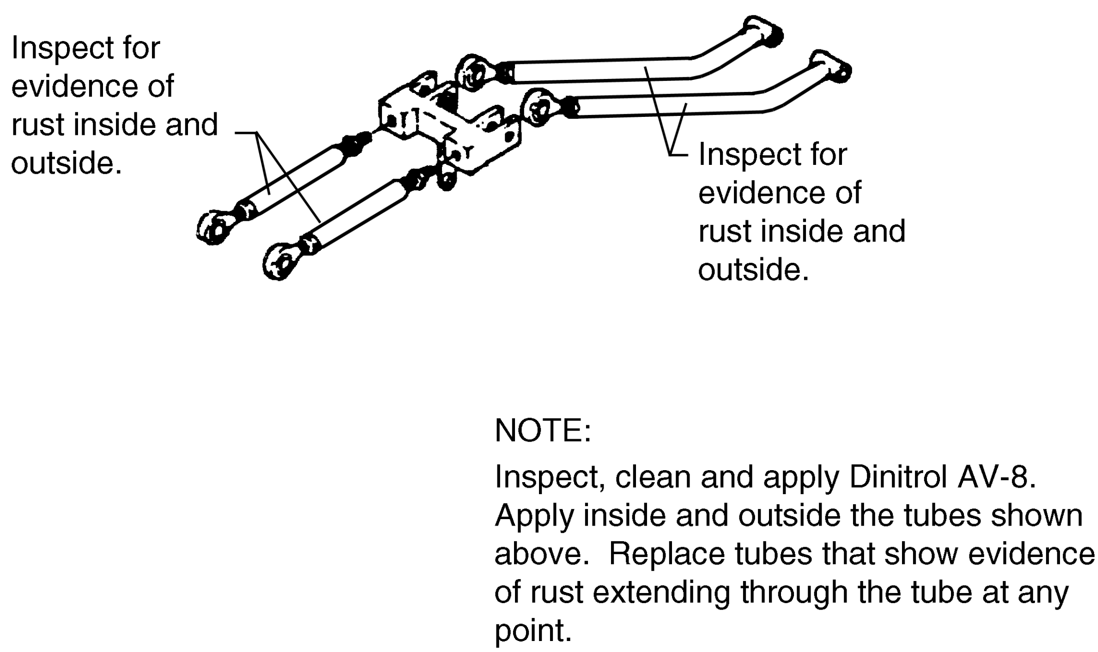
A. When inspection and repair or replacement are completed, apply Dinitrol AV 8 to elevator trim tab actuator push rod tubes.

CAUTION: READ PRECAUTIONS ON CONTAINER LABEL AND WEAR A CHARCOAL FILTER MASK WHEN APPLYING DINITROL AV 8.

B. Apply Dinitrol by sprayer or brush. Use an air or airless spray gun, if available. A hand aspirator bottle may be used if a sufficient fan spray pattern can be developed. Product should not be thinned.

C. Spray directly into mating points. Pour inside the lower (curved) elevator trim tab actuator push rod tubes until full and drain.

D. Apply to approximately .8 mils in thickness. Avoid puddling and running. Allow to dry (6 to 8 hours) before reinstalling elevator trim tab rods.



Elevator Trim Tab Rods
Figure 3

5. Elevator Plate Assembly Rivet Inspection

NOTE: This inspection incorporates the requirements of Piper Service Letter No. 1183.

Each100 hours time-in-service, conduct a visual inspection of the seven (7) rivets attaching the elevator plate assembly, Piper Part Number (P/N) 83515-002 (also known as an elevator control horn) to the elevator rib P/N 83508-002. Use a borescope and/or a 10X magnifier, a mirror and a suitable light source to conduct inspection.

A. Remove the aft fuselage fairing strap, shown in “Figure 4” on page 55207.

CAUTION: ANY DAMAGED, MISSING OR SMOKING RIVETS MUST BE REPLACED PRIOR TO NEXT FLIGHT. RIVET REPLACEMENT DOES NOT RELIEVE THE RECURRING INSPECTION REQUIREMENT FOR ANY RIVETS.

B. Inspect all seven rivet locations, as identified in “Figure 4” on page 55208 for damaged, missing or smoking rivets, and replace on condition. Any parts with damage that exceeds the repair limitations must be replaced.

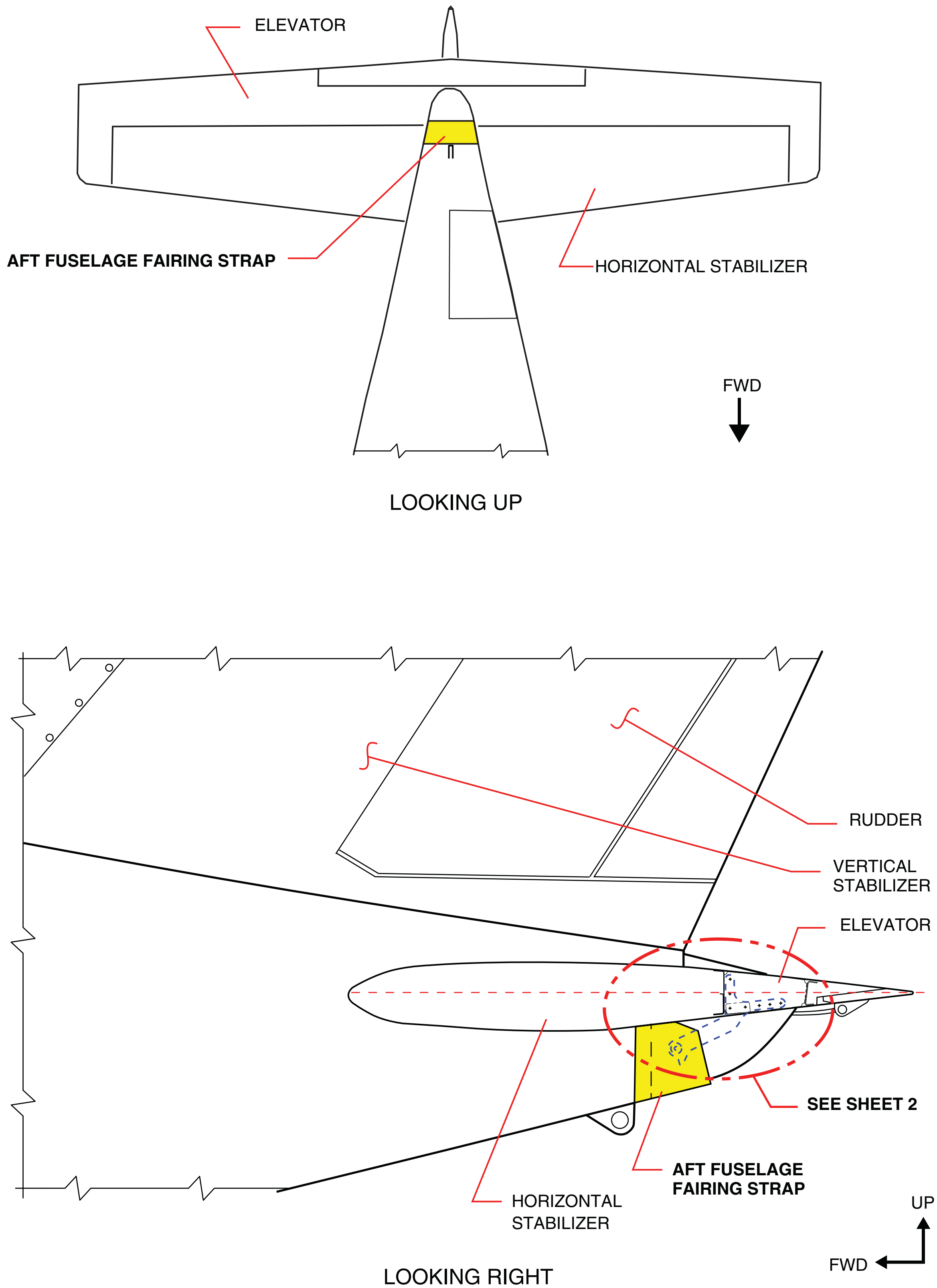
NOTE: The term “smoking rivet” is used to describe a loose or working rivet whose vibration often creates a black streak trailing aft.

C. When replacing rivets in airframe structure, refer to FAA Advisory Circular AC 43.13-1B (Acceptable Methods, Techniques, and Practices - Aircraft Inspection and Repair), Chapter 4, Section 4, for further guidance.

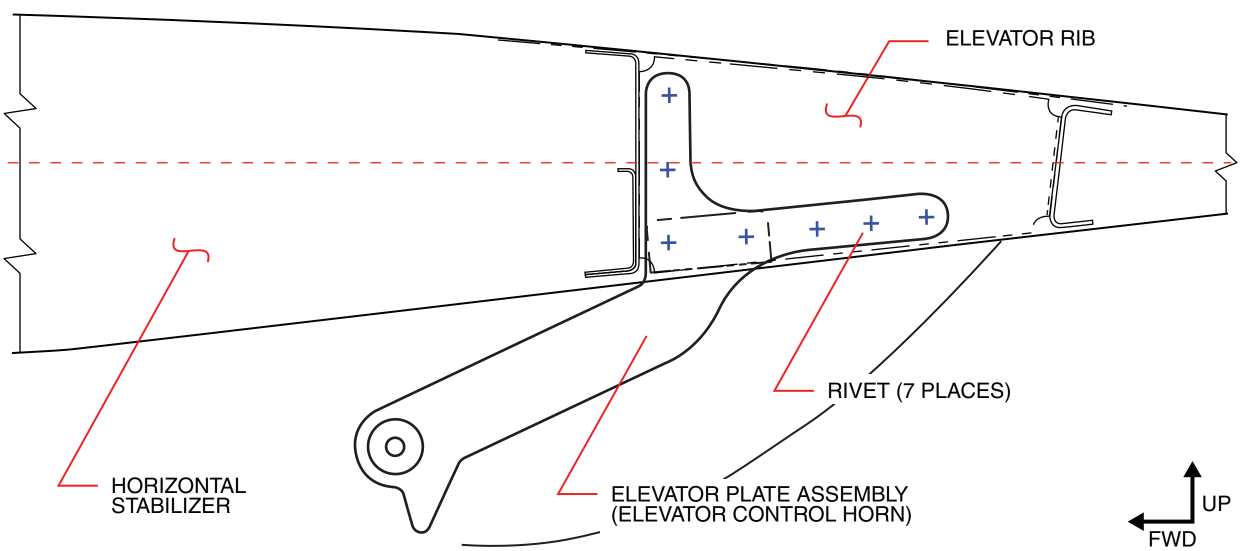
D. Reinstall the aft fuselage fairing strap.

E. Make a logbook entry documenting compliance with this service letter.

NOTE: Report to Piper any damage requiring the repairs specified herein, at (1)(772) 299-2141 or email customer.service@piper.com. Piper’s normal business hours are 7:30 a.m. to 4:30 p.m. (Eastern U.S.), Monday through Friday.



Elevator Plate Assembly Rivet Inspection
Figure 4 (Sheet 1 of 2)



NOTE: SOME COMPONENTS REMOVED FOR CLARITY.

LOOKING RIGHT

Elevator Plate Assembly Rivet Inspection
Figure 4 (Sheet 2 of 2)

VERTICAL STABILIZER

1. Vertical Fin

See “Figure 1”, 55-00-00.

A. Removal

(1) Remove the tailcone access panel.

(2) Remove the dorsal fairing.

(3) Remove the rudder per instructions in this chapter.

(4) Remove the forward vertical fin attaching bolt.

(5) Support the fin and remove the aft vertical fin attaching nuts, bolts and washers. Remove the fin from the airplane.

B. Installation

(1) Position the vertical fin on the airplane.

(2) Attach the aft vertical fin spar to the fuselage spar with the bolts, nuts and washers. Torque from 160 to 200 in. lbs.

(3) Attach the forward vertical fin spar to the fuselage spar. Torque the bolt as specified in “Figure 1”, 55-00-00.

(4) Attach the rudder to the vertical fin per instructions in this chapter.

(5) Check the rigging of the rudder (refer to 27-20-00).

(6) Install the dorsal fairing (seal fairing to skin and fairing recess angle using Bostik Chem-Calk® Urethane No. 915 - remove excess sealant) and the tailcone access panel.

2. Attach Fitting Corrosion Control

Each five years, inspect the attach fittings shown in “Figure 1” on page 55302 for corrosion and condition. Repair/replace as required. When inspection and repair/replacement is completed, reapply Dinitrol AV 8 (P/N 89500-800) Corrosion Inhibiting Compound.

NOTE: For all fittings, inspection of the exposed portion is sufficient unless obvious or extensive corrosion is apparent.

NOTE: Any part exhibiting flaking of the metal due to rust must be replaced.

NOTE: If aluminum skin or structure is corroded (beyond a light surface oxidation), the area must be repaired or replaced.

NOTE: On any part being replaced, dip rivets in fluid resistant epoxy primer or equivalent and install wet.

A. When inspection and repair or replacement are completed, allow sufficient time for primer to dry, and apply Dinitrol AV 8 to forward vertical fin attach point.

CAUTION: READ PRECAUTIONS ON CONTAINER LABEL AND WEAR A CHARCOAL FILTER MASK WHEN APPLYING DINITROL AV 8.

B. Apply Dinitrol by sprayer or brush. Use an air or airless spray gun, if available. A hand aspirator bottle may be used if a sufficient fan spray pattern can be developed. Product should not be thinned.

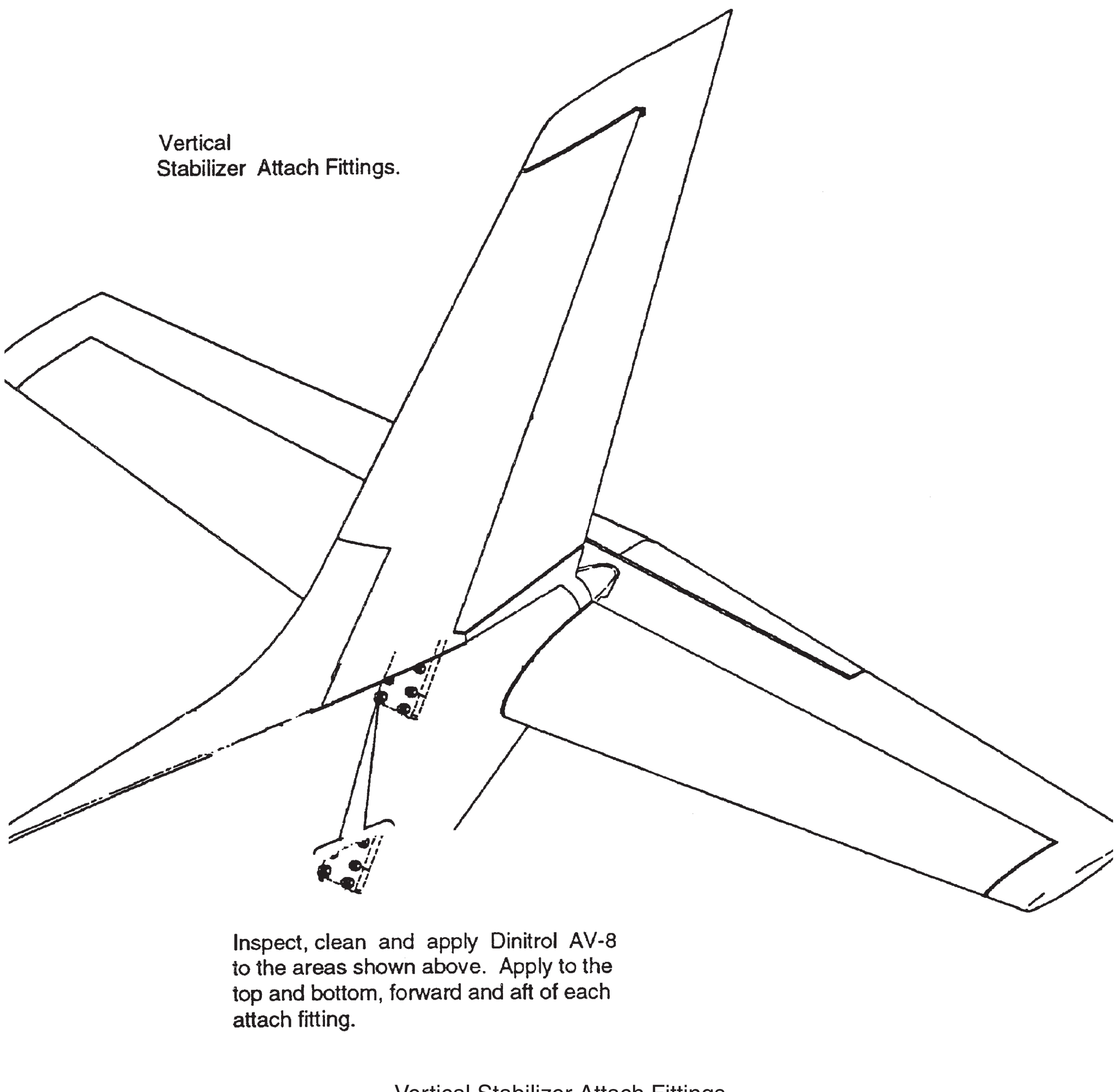
C. Cover safety and outflow valves, air conditioning condenser, and other components on the pressure bulkhead. Cover autopilot servos, electrical connectors, and cable pulleys.

WARNING: DO NOT GET DINITROL INTO AUTOPILOT SERVOS OR CLUTCHES.

D. Spray directly into faying surfaces and mating points.

E. Ensure entire attach fitting is treated. Spray inside where the fitting is attached to the structure.

F. Apply to approximately .8 mils in thickness. Avoid puddling and running. Allow to dry 1 to 1 1/2 hours before airplane use. Full drying takes 6 to 8 hours.



Vertical Stabilizer Attach Fittings
Figure 1

RUDDER

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

Rudder

See “Figure 1” on page 55001.

A. Removal

(1) Remove the tailcone access panel.

(2) Remove the rudder cables from the rudder sector, or loosen them if not removing the sector.

(3) Disconnect the sector from the rudder torque tube by removing 3 bolts, nuts and washers.

(4) Remove the sector retaining nut, bolt and washers, if sector is to be removed.

(5) Support the rudder and remove the two sets of rudder hinge retaining bolts, nuts and washers. Remove the rudder from the airplane.

B. Installation

WARNING: IF THE RUDDER HAS BEEN REPLACED OR REPAINTED, THE RUDDER MUST BE BALANCED BEFORE INSTALLATION. SEE BALANCING, BELOW.

(1) Position the rudder on the airplane.

(2) Install the rudder hinge bolts and washers. Adjust washers above and below to achieve minimum gap at bottom of rudder as shown in ”Figure 1” on page 55003. Torque the nut as specified in “Figure 1” on page 55001.

(3) If previously removed, install the rudder sector, securing it with bolt, nut and washers. Torque the nut as specified in “Figure 1” on page 55001.

(4) Attach the rudder sector to the rudder torque tube with three (3) bolts, nuts and washers. Torque the nuts as specified in “Figure 1” on page 55001.

(5) Attach the rudder cables to the sector (if the cables were previously removed).

(6) Set tension on cables per 27-20-00.

(7) Check rigging and operation of rudder control system per 27-20-00.

(8) Install the tailcone access panel.

C. Balancing

See “Figure 1” on page 55402.

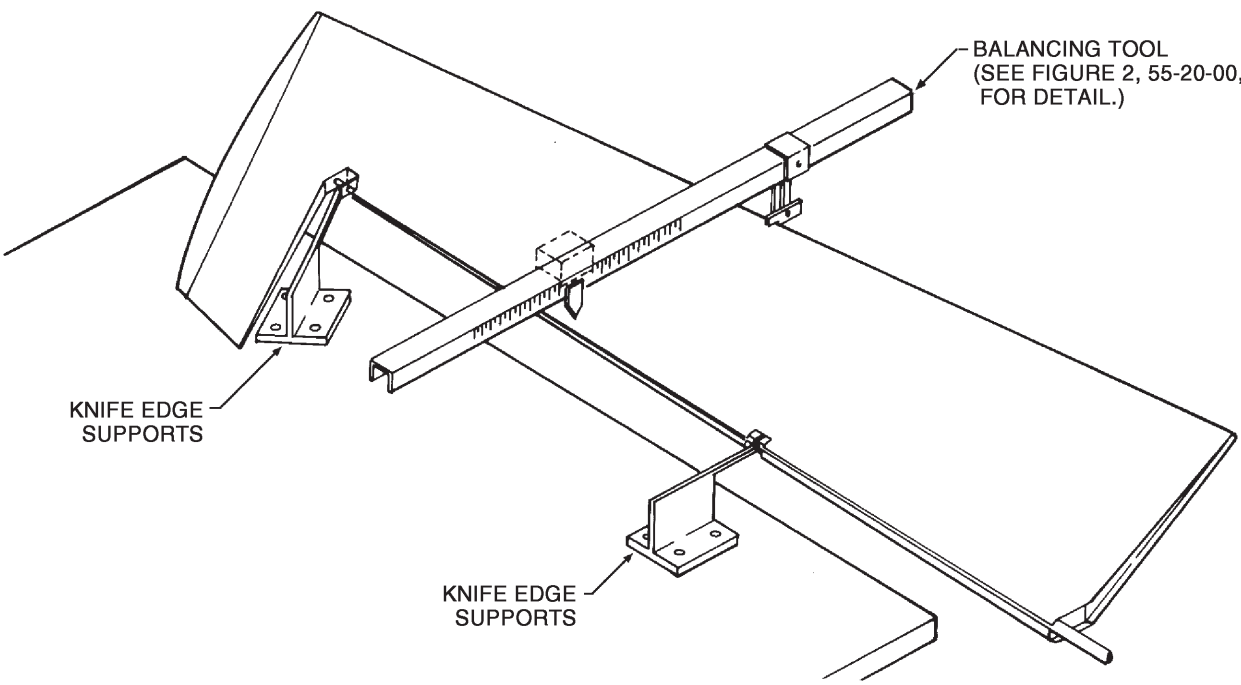
To balance the rudder, the assembly must be complete and in its final flight configuration as specified under “Balancing Equipment” on page 55202. Place the complete assembly horizontally on the knife edge support of a balancer. Ensure that there are no drafts in the balancing room. Place the tool on the rudder with the beam perpendicular to the hinge centerline. Calibrate the tool per “Balancing Equipment” on page 55202. Read the scale when the bubble level has been centered by adjustment of the movable weight and determine the static balance limit. If the static balance is not within the limits given in “Chart 1” on page 55402, proceed as follows:

(1) Nose Heavy: This condition is highly improbable: recheck calculations and measurements.

(2) Nose Light: In this case, the rudder is too heavy because of painting or repairs; it will be necessary to strip the paint and repaint the rudder. If the rudder is too heavy as a result of repairs, the repair must be removed and the damaged parts replaced.

CHART 1
RUDDER STATIC BALANCE LIMITS

Leading Edge Heavy (Inch-Pounds)	Trailing Edge Heavy (Inch-Pounds)
0	-32.0



Balancing Rudder
Figure 1

CHAPTER

56

WINDOWS

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CHAPTER 56

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
56-List of Effective Pages	1	Dec 15/14			
	2	Sep 15/09			
56-Table of Contents	1	Dec 15/14			
	2	Sep 15/09			
56-00-00	1	Sep 15/09			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Sep 15/09			
	5	Sep 15/09			
	6	Sep 15/09			
	7	Sep 15/09			
	8	Sep 15/09			
	9	Sep 15/09			
	10	Sep 15/09			
	11	Sep 15/09			
	12	Sep 15/09			
	13	Jan 15/13			
	14	Sep 15/09			
56-10-00	1	Jan 15/13			
	2	Sep 15/09			
	3	Dec 15/14			
	4	Jan 15/13			
	5	Jan 15/13			
	6	Jan 15/13			
	7	Jan 15/13			
	8	Jan 15/13			
	9	Jan 15/13			
	10	Sep 15/09			
56-20-00	1	Dec 15/14			
	2	Sep 15/09			
56-30-00	1	Dec 15/14			
	2	Sep 15/09			

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CHAPTER 56 - WINDOWS

TABLE OF CONTENTS

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GENERAL

The airplane is equipped with a two piece windshield and eight cabin windows (one is in the door and one is the emergency exit window).

The pilot's heated windshield is made of two-ply, laminated glass (PA-46-350P - Standard & PA-46R-350T Optional). On S/N's 4692001 and up, the standard pilot's non-heated windshield is made of clear acrylic.

The co-pilot's windshield and cabin windows are either made of clear stretched acrylic (PA-46-350P) or clear acrylic (PA-46R-350T). They are installed with sealant and fasteners as necessary to retain them and prevent water or air leaks.

1. Limitations

   A. Flight with cracked or crazed heated glass windshield is prohibited.

   B. Flight with cracked or crazed acrylic windshield/window is prohibited except as follows:

      (1) Flight with cracked windshield/window prohibited except as follows:

         ✈ Upon FAA-accepted repair (i.e., AC 43.13-1 latest revision) of the windshield/window, unpressurized flight is permitted pending installation of replacement windshield/window;

         ✈ Crack(s) emanate solely from bolt holes and are within the limits prescribed for bolt holes in "Crack Inspection - Acrylic Windows and Windshields," below.

      (2) Flight with crazed windshield/window prohibited except as follows:

         Unpressurized flight is permitted solely to return to home base and/or reposition to maintanence facility. No other flight is authorized.

   C. Pressurized flights with discoloration or haze should be avoided but, if absolutely necessary while awaiting replacement, flights may be made if careful inspection for crazing is implemented before each flight.

2. Definitions

   The following apply to all cockpit and cabin windows and windshields.

Critical	The viewing area of the windshields used for taxiing, takeoff, climb, cruise, and landing.
Semi-Critical	The viewing area used for general flight vision.
Non-Critical	Viewing areas not normally used for flight operations.
Distortion	Lines in windows or windshields that cause waviness in objects when looking through the window or windshield.
Crack	Critical narrow break, fissure, or separation extending through the entire thickness of the transparent material.
Craze	Fissure on the surface of the transparent material that does not penetrate the full thickness of the material.
Crazing	Mesh of fine hairline cracks that do not penetrate the full thickness of the material, located on the surface or within the structure of the transparent material.
Star Craze	A condition where several fissures radiate from a central point.
Wedge	A condition in a piece of optical glass having a progressive variation in thickness or absorption from one side to the other.
Scratch	An abrasion on the surface of the material caused by contact with rough abrasives or sharp objects.
Hairline Scratch	Visible scratch undetectable when passing a fingernail over the scratch. Considered non-critical other than being an appearance defect.

Light Scratch	A scratch measuring less than 0.010 inch (0.254 mm) deep. Can be detected when passing a fingernail over the scratch. Considered non-critical except for appearance.
Heavy Scratch	A scratch measuring more than 0.010 inch (0.254 mm) deep. Can be detected when passing a fingernail over the scratch. This type of scratch may be accompanied by chipping along the edge. Considered critical when occurring to inner glass ply. Considered semi-critical when occurring to outer glass ply within certain limitations (see inspection criteria). Considered non-critical within certain limitations (see inspection criteria) when occurring to acrylic surfaces.
Chip	A chip is considered a small scratch.
Peel Chip	Peel chips, also known as adhesion chips or cold chips, occur on the laminated surface of a glass ply when the vinyl interlayer pulls a chip of glass out of the glass ply.
Haze	A foggy appearance located on the surface of the transparent material.
Blemish	Speck, air bubble, or other minor imperfection imbedded in the transparent material.
Mark-Off	An almost nonexistent shallow depression on the surface possessing practically no depth. Discernible only due to a noticeable rim or roughened surface caused by mold surface defects transferred to the surface during the forming operation.
Delamination	Visible evidence of a physical break of the bond between the plastic interlayer and either glass ply. Delamination may be caused by laminating stresses, preload on installation, or excessive heat. Cloudy or milky appearance in the delamination indicates moisture or solvent penetration.

3. Inspection and Repair - (PA-46-350P ONLY)

CAUTION: DAMAGE LIMITS AND REPAIRS STATED HEREIN MUST BE APPLIED CONSERVATIVELY. REWORK OF MULTIPLE CHIPS OR SCRATCHES SHOULD NOT BE ATTEMPTED IN ANY CASE WHERE, WHEN FINISHED, A LINEAR PATTERN WILL HAVE BEEN CREATED.

A. Heated Glass Windshield (Pilot's Side) Inspection

CAUTION: FLIGHT WITH CRACKED OR CRAZED HEATED WINDSHIELD PROHIBITED.

The pilot's windshield is a two-ply laminated glass containing a heating element.

(1) Critical Area Inspection. The area of the laminated glass windshield used for taxiing, takeoff, climb, cruise, and landing.

(a) Cracks can occur in either the inner, outer, or both glass plies. Cracking in either ply is critical and cause for immediate replacement.

(b) Crazing.

1) Crazing in the windshield is critical.

2) Determine depth of craze using a calibrated depth gauge or scale (see “Figure 1”).

3) A craze of 0.062 inch (1.575 mm) depth is cause for immediate replacement. A craze of 0.031 inch (0.787 mm) depth is cause for replacement at the earliest opportunity.

4) Crazing in any portion of the windshield requires replacement.

(c) Blemishes.

1) Blemishes in windshields form in the vinyl plastic interlayer bonding the two glass plies together.

2) Blemishes in the critical or semi-critical portion of the windshield are not acceptable unless 0.062 inch (1.575 mm) or smaller in circumference, including distorted area.

3) No more than two blemishes, at least 12 inches (30.48 cm) apart are acceptable in the windshield.

4) No more than two blemishes within a two inch (5.08 cm) area along the upper portion of the windshield are acceptable.

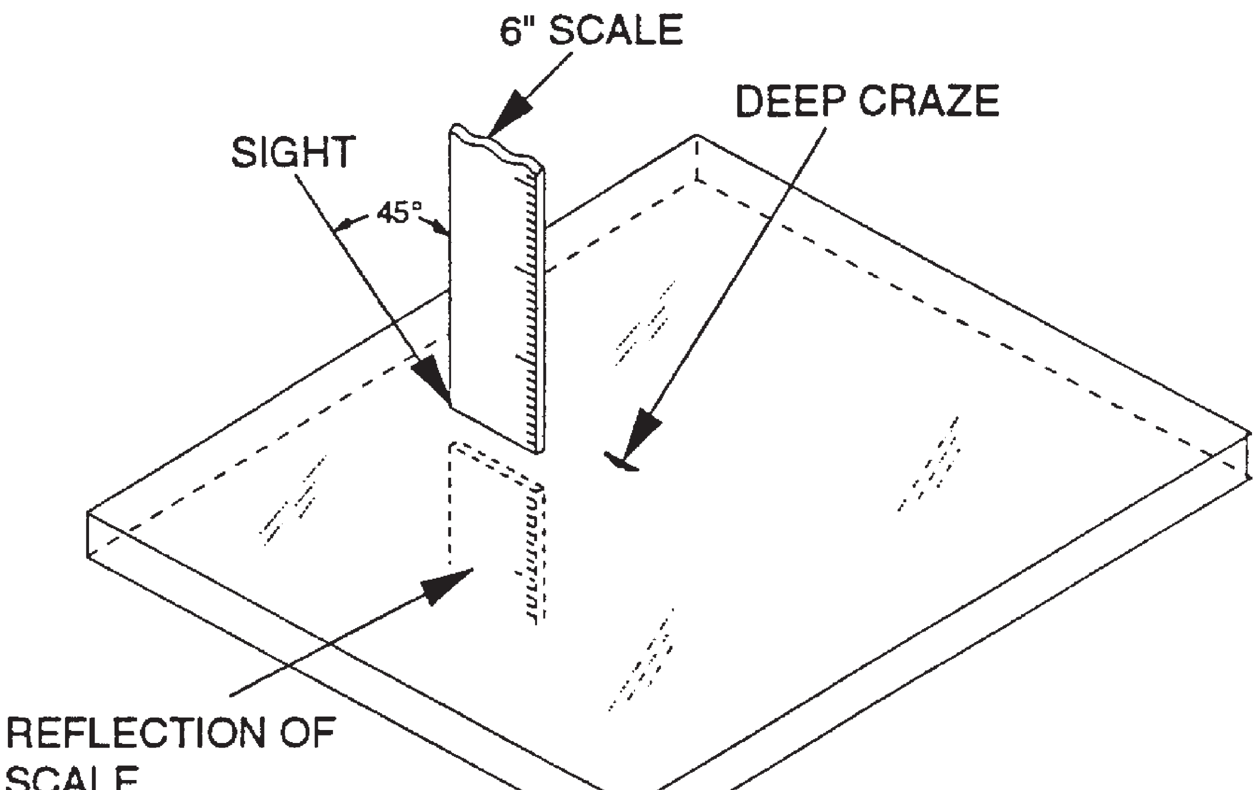
(d) Haze or foggy appearance on the glass is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.

(e) Scratches.

1) Scratches in the outboard surface of the outer glass ply no longer than 0.062 inch (1.575 mm) and no deeper than 0.020 inch (0.508 mm) are allowable.

2) Scratches in the inboard surface of the inner glass ply with a depth less than or equal to 0.002 inch (0.05 mm) are allowable.

3) Scratches which obstruct vision required for safe operation of the aircraft or exceed the above limits are cause for immediate windshield replacement.



Determining Depth of Craze
Figure 1

(f) Delamination.

Cloudy or milky appearance in the critical area (i.e. - obstructing vision required for safe operation of the aircraft) is cause for immediate replacement.

(g) Peel Chips.

1) Areas characterized by irregular or jagged boundaries indicate uneven separation of the vinyl and glass. This condition may cause the vinyl to pull chips from the inner glass ply, resulting in failure of the glass ply. Conduct periodic inspections to determine if the damage is progressive or if chipping of the inner glass ply is present.

2) Replace windshield if any of the following conditions are noted:

a) Evidence of chipping of inner glass ply.

b) Area of vision required for safe operation is affected.

c) Windshield heat system inoperative.

(h) Mark-Off of such low intensity that vision quality is not impaired and that is not visible when looking through the windshield is acceptable.

(i) Distortion. Slight horizontal distortion is acceptable, if: there are no more than two lines; they do not occupy more than 25 percent of the windshield area, and; they are separated by a minimum of six inches (15.24 cm).

(2) Semi-Critical Area Inspection. The perimeter area of the heated glass windshield used for general flight vision.

(a) Blemishes.

1) Blemishes in windshields form in the vinyl plastic interlayer bonding the two glass plies together.

2) Blemishes in the critical or semi-critical portion of the windshield are not acceptable unless 0.062 inch (1.575 mm) or smaller in circumference, including distorted area.

3) No more than two blemishes, at least 12 inches (30.48 cm) apart are acceptable in the windshield.

4) No more than two blemishes within a two inch (5.08 cm) area along the upper portion of the windshield are acceptable.

(b) Haze or foggy appearance on the glass is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.

(c) Crazing.

1) Crazing in the windshield is critical.

2) Determine depth of craze using a calibrated depth gauge or scale (see “Figure 1”).

3) A craze 0.031 inch (0.787 mm) deep is cause for replacement at the earliest opportunity.

4) Crazing in any portion of the windshield requires replacement.

(d) Scratches.

1) Scratches in the outboard surface of the outer glass ply no longer than 0.062 inch (1.575 mm) and no deeper than 0.020 inch (0.508 mm) are allowable.

2) Scratches in the inboard surface of the inner glass ply with a depth less than or equal to 0.002 inch (0.05 mm) are allowable.

3) Scratches which obstruct vision required for safe operation of the aircraft or exceed the above limits are cause for immediate windshield replacement.

(e) Delamination.

1) Delamination is allowable up to two inches along the perimeter of the daylight opening of the windshield. In the corners, up to four inches is allowable.

2) Windshields exhibiting delamination must be monitored as the condition is usually progressive.

3) Replace windshield if any of the following condition are noted:

d) Delamination exceeds allowable limits.

e) Area of vision required for safe operation is affected.

a) Windshield heat system inoperative.

(f) Peel Chips.

1) Delaminated areas characterized by irregular or jagged boundaries indicate uneven separation of the vinyl and glass. This condition may cause the vinyl to pull chips from the inner glass ply, resulting in failure of the glass ply. Conduct periodic inspections to determine if the damage is progressive or if chipping of the inner glass ply is present.

2) Replace windshield if any of the following condition are noted:

a) Evidence of chipping of inner glass ply.

b) Area of vision required for safe operation is affected.

c) Windshield heat system inoperative.

(g) Mark-Off of such low intensity that vision quality is not impaired and that is not visible when looking through the windshield is acceptable.

(h) Distortion.

1) A moderate amount of distortion is acceptable along the lower portion of the windshield if no more than 1.5 inches (3.81 cm) above the windshield retainer strip.

2) Distortion may not be so severe that it restricts vision or diverts runway lines or section lines more than 45°.

3) Slight horizontal distortion is acceptable, if: there are no more than two lines; they do not occupy more than 25 percent of the windshield area, and; they are separated by a minimum of six inches (15.24 cm).

B. Stretched Acrylic Windshield (Co-Pilot's Side) Inspection

CAUTION: FLIGHT WITH CRACKED WINDSHIELD PROHIBITED EXCEPT AS FOLLOWS:

- UPON FAA-ACCEPTED REPAIR (I.E., AC 43.13-1 LATEST REVISION) OF THE WINDSHIELD, UNPRESSURIZED FLIGHT IS PERMITTED PENDING INSTALLATION OF REPLACEMENT WINDSHIELD;

- CRACK(S) EMANATE SOLELY FROM BOLT HOLES AND ARE WITHIN THE LIMITS PRESCRIBED FOR BOLT HOLES IN "CRACK INSPECTION - ACRYLIC WINDOWS AND WINDSHIELDS," BELOW.

CAUTION: FLIGHT WITH CRAZED WINDSHIELD PROHIBITED EXCEPT AS FOLLOWS: UNPRESSURIZED FLIGHT IS PERMITTED SOLELY TO RETURN TO HOME BASE AND/ OR REPOSITION TO MAINTANENCE FACILITY. NO OTHER FLIGHT IS AUTHORIZED.

The co-pilot's windshield is formed from 0.312 inch thick stretched acrylic.

(1) Critical Area Inspection. The area of the windshield used for taxiing, takeoff, climb, cruise, and landing.

(a) Cracking in the windshield is critical and cause for immediate replacement. Inspect per Crack Inspection - Acrylic Windows and Windshields, below.

(b) Crazing.

1) Crazing in the windshield is critical.

2) Determine depth of craze using a calibrated depth gauge or scale (see “Figure 1”).

3) A craze of 0.062 inch (1.575 mm) depth is cause for immediate replacement. A craze of 0.031 inch (0.787 mm) depth is cause for replacement at the earliest opportunity.

4) Crazing in any portion of the windshield requires replacement.

(c) Haze or foggy appearance is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.

(d) Scratches.

1) Limits:

1.5 inch (38.1 mm) long

0.031 inch (0.787 mm) deep

0.2 inch (5.080 mm) wide

0.25 in^2 (1.613 cm^2) total area

2) No more than two allowed per 10 in^2 (64.516 cm^2) of area (i.e. - an area approximately 3.16 inch by 3.16 inch).

3) Scratches exceeding these limits are cause for immediate windshield replacement / rework (see Window / Windshield Repair / Rework, below).

(e) Mark-Off of such low intensity that vision quality is not impaired and that is not visible when looking through the windshield is acceptable.

(f) Distortion. Slight horizontal distortion is acceptable, if: there are no more than two lines; they do not occupy more than 25 percent of the windshield area, and; they are separated by a minimum of six inches (15.24 cm).

(2) Semi-Critical Area Inspection. The perimeter area of the acrylic windshield used for general flight vision.

(a) Crazing.

1) Crazing in the windshield is critical.

2) Determine depth of craze using a calibrated depth gauge or scale (see “Figure 1”).

3) A craze 0.031 inch (0.787 mm) deep is cause for replacement at the earliest opportunity.

4) Crazing in any portion of the windshield requires replacement.

(b) Haze or foggy appearance is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.

(c) Scratches.

1) Scratches in acrylic windshields may be reworked per Window / Windshield Repair / Rework, below, if:

a) They are less than 0.030 inch (0.762 mm) deep.

b) Windshield is a minimum of 0.279 inch (7.087 mm) thick after rework.

c) No vision distortions in critical and semi-critical areas as a result of rework.

2) Replace acrylic windshields with scratches that cannot be reworked in accordance with these standards.

(d) Mark-Off of such low intensity that vision quality is not impaired and that is not visible when looking through the windshield is acceptable.

(e) Distortion.

1) A moderate amount of distortion is acceptable along the lower portion of the windshield if no more than 1.5 inches (3.81 cm) above the windshield retainer strip.

2) Distortion may not be so severe that it restricts vision or diverts runway lines or section lines more than 45°.

3) Slight horizontal distortion is acceptable, if: there are no more than two lines; they do not occupy more than 25 percent of the windshield area, and; they are separated by a minimum of six inches (15.24 cm).

C. Stretched Acrylic Window (Cockpit) Inspection

CAUTION: FLIGHT WITH CRACKED WINDOW PROHIBITED EXCEPT AS FOLLOWS:

- UPON FAA-ACCEPTED REPAIR (I.E., AC 43.13-1 LATEST REVISION) OF THE WINDOW, UNPRESSURIZED FLIGHT IS PERMITTED PENDING INSTALLATION OF REPLACEMENT WINDOW;
- CRACK(S) EMANATE SOLELY FROM BOLT HOLES AND ARE WITHIN THE LIMITS PRESCRIBED FOR BOLT HOLES IN “CRACK INSPECTION - ACRYLIC WINDOWS AND WINDSHIELDS,” BELOW.

CAUTION: FLIGHT WITH CRAZED WINDOW PROHIBITED EXCEPT AS FOLLOWS: UNPRESSURIZED FLIGHT IS PERMITTED SOLELY TO RETURN TO HOME BASE AND/OR REPOSITION TO MAINTANENCE FACILITY. NO OTHER FLIGHT IS AUTHORIZED.

(1) The cockpit windows are formed from 0.250 inch thick stretched acrylic.

(2) Semi-Critical Area Inspection. The areas of cockpit acrylic windows used for general flight vision.

(a) Cracks.

Inspect for cracks per Crack Inspection - Acrylic Windows and Windshields, below.

(b) Crazing.

1) Crazing in the windows is critical.

2) Determine depth of craze using a calibrated depth gauge or scale (see “Figure 1”).

3) A craze 0.050 inch (1.270 mm) deep is cause for immediate replacement. A craze 0.025 inch (0.635 mm) deep is cause for replacement at the earliest opportunity.

4) Crazing in any portion of a window requires replacement.

(c) Haze or foggy appearance is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.

(d) Scratches.

1) Limits:

1.5 inch (38.1 mm) long

0.025 inch (0.635 mm) deep

0.2 inch (5.080 mm) wide

0.25 in^2 (1.613 cm^2) total area

2) No more than two allowed per 10 in^2 (64.516 cm^2) of area (i.e. - an area approximately 3.16 inch by 3.16 inch).

3) Scratches exceeding these limits are cause for immediate windshield replacement / rework (see Window / Windshield Repair / Rework, below).

4) Scratches in acrylic cockpit windows may be reworked per Window / Windshield Repair / Rework, below, if:
   a) They are less than 0.025 inch (0.635 mm) deep.
   b) Window is a minimum of 0.225 inch (5.715 mm) thick after rework.
   c) No vision distortions in critical and semi-critical areas as a result of rework.
5) Replace acrylic windows with scratches that cannot be reworked in accordance with these standards.

(e) Mark-Off of such low intensity that vision quality is not impaired and that is not visible when looking through the windshield is acceptable.

(f) Distortion.
1) A moderate amount of distortion is acceptable along the lower portion of the windshield if no more than 1.5 inches (3.81 cm) above the windshield retrainer strip.
2) Distortion may not be so severe that it restricts vision or diverts runway lines or section lines more than 45°.
3) Slight horizontal distortion is acceptable, if: there are no more than two lines; they do not occupy more than 25 percent of the windshield area, and; they are separated by a minimum of six inches (15.24 cm).

D. Stretched Acrylic Window (Cabin) Inspection

CAUTION: FLIGHT WITH CRACKED WINDOW PROHIBITED EXCEPT AS FOLLOWS:
- UPON FAA-ACCEPTED REPAIR (I.E., AC 43.13-1 LATEST REVISION) OF THE WINDOW, UNPRESSURIZED FLIGHT IS PERMITTED PENDING INSTALLATION OF REPLACEMENT WINDOW;
- CRACK(S) EMANATE SOLELY FROM BOLT HOLES AND ARE WITHIN THE LIMITS PRESCRIBED FOR BOLT HOLES IN “CRACK INSPECTION - ACRYLIC WINDOWS AND WINDSHIELDS,” BELOW.

CAUTION: FLIGHT WITH CRAZED WINDOW PROHIBITED EXCEPT AS FOLLOWS: UNPRESSURIZED FLIGHT IS PERMITTED SOLELY TO RETURN TO HOME BASE AND/ OR REPOSITION TO MAINTANENCE FACILITY. NO OTHER FLIGHT IS AUTHORIZED

(1) The cabin windows are formed from 0.210 inch thick acrylic.
(2) Non-Critical Area Inspection. Any portion of a window not used for flight vision.
(a) Cracks.
Inspect for cracks per Crack Inspection - Acrylic Windows and Windshields, below.
(b) Crazing.
1) Crazing in the windows is critical.
2) Determine depth of craze using a calibrated depth gauge or scale (see “Figure 1”).
3) A craze of 0.043 inch (1.092 mm) depth is cause for immediate replacement. A craze 0.021 inch (0.533 mm) deep is cause for replacement at the earliest opportunity.
4) Crazing in any portion of a window requires replacement.
(c) Haze or foggy appearance on the glass is not acceptable if the amount of haze/fog causes an obstruction of vision in the area used for operation of the aircraft.
(d) Mark-Off in moderate amounts is acceptable provided that visibility is not impaired.
(e) Distortion.
1) Distortion along the sides of the window within 0.5 inches (12.70 mm) or less of the retainer is acceptable.

2) Distortion along the upper portion of the window is acceptable if distortion does not extend downward more than 2.00 inches (50.8 mm) from the top and does not impair vision.

3) Distortion of the main body of the side window is acceptable providing the distorted area does not exceed 25 percent of the window area.

E. Window / Windshield Repair / Rework Procedure

(Stretched Acrylic Windows / Co-Pilot's Stretched Acrylic Windshield). The following methods should be used for repairs:

CAUTION: ANY ACRYLIC WINDOW WHICH MAY HAVE BEEN DAMAGED BY PAINT THINNER, PAINT REMOVER OR OTHER SOFTENING AGENT MUST BE REPLACED. NO REPAIR IS PERMITTED FOR THIS TYPE OF DAMAGE.

CAUTION: DAMAGE LIMITS AND REPAIRS STATED HEREIN MUST BE APPLIED CONSERVATIVELY. REWORK OF MULTIPLE CHIPS OR SCRATCHES SHOULD NOT BE ATTEMPTED IN ANY CASE WHERE, WHEN FINISHED, A LINEAR PATTERN WILL HAVE BEEN CREATED.

Rework the damaged area to the depth of the damage, approximately twice the length and three times the width, removing all sharp edges.

(1) Areas with small scratches:
  (a) Clean the window, using generous amounts of water and a mild detergent.
  (b) Polish the window with an approved compound and soft cloth.
  (c) Clean and wax the polished area.

(2) Areas with large scratches, gouges, and nicks: Areas with damages exceeding .003 of an inch depth or those with less than .003 of an inch in depth, having sharp edges which cause hanging of fingernail should be locally rounded out or buffed.
  (a) Clean the window using generous amounts of water and a mild detergent.
  (b) Use a scratch removal kit, such as the type supplied by Micro-Surface Finishing Products Inc., P.O. Box 456, Wilton, Iowa, to remove the defective area, blend and buff.
  (c) Using 400A wet or dry abrasive paper wrapped around a smooth rubber block and generous amounts of water, lightly sand over and around the defected area in a circular motion, extending in a diameter equal to two or three times the defected area.
  (d) Continue sanding until the initial defect is no longer apparent. Thoroughly flush the area with water.
  (e) Using 600A wet or dry abrasive paper, repeat step (c). Continue sanding only until the hairline scratches caused by the coarse sanding are no longer apparent. Sand a larger area than that covered by the original sanding operation. Thoroughly wash the area.
  (f) Finish the repair using instructions given in paragraph (1), above.

4. Inspection and Repair - (PA-46R-350T ONLY)

CAUTION: DAMAGE LIMITS AND REPAIRS STATED HEREIN MUST BE APPLIED CONSERVATIVELY. REWORK OF MULTIPLE CHIPS OR SCRATCHES SHOULD NOT BE ATTEMPTED IN ANY CASE WHERE, WHEN FINISHED, A LINEAR PATTERN WILL HAVE BEEN CREATED.

A. Heated Glass Windshield (Pilot's Side - Optional) Inspection

CAUTION: FLIGHT WITH CRACKED OR CRAZED HEATED WINDSHIELD PROHIBITED.

The pilot's windshield is a two-ply laminated glass containing a heating element. See procedures located in the "Inspection and Repair - (PA-46-350P ONLY)" section above.

B. Windshield / Window Repair / Rework Procedure (Acrylic Windshield / Window).

CAUTION: FLIGHT WITH CRACKED ACRYLIC WINDSHIELD/WINDOW PROHIBITED EXCEPT AS FOLLOWS:

- UPON FAA-ACCEPTED REPAIR (I.E., AC 43.13-1 LATEST REVISION) OF THE WINDOW, UNPRESSURIZED FLIGHT IS PERMITTED PENDING INSTALLATION OF REPLACEMENT WINDOW;
- CRACK(S) EMANATE SOLELY FROM BOLT HOLES AND ARE WITHIN THE LIMITS PRESCRIBED FOR BOLT HOLES IN “CRACK INSPECTION - ACRYLIC WINDOWS AND WINDSHIELDS,” BELOW.

CAUTION: FLIGHT WITH CRAZED ACRYLIC WINDSHIELD/WINDOW PROHIBITED EXCEPT AS FOLLOWS:

FLIGHT IS PERMITTED SOLELY TO RETURN TO HOME BASE AND/OR REPOSITION TO MAINTANENCE FACILITY. NO OTHER FLIGHT IS AUTHORIZED.

See AC 43.13-1.

5. Crack Inspection - Acrylic Windows and Windshields

CAUTION: FLIGHT WITH CRACKED WINDSHIELD/WINDOW PROHIBITED EXCEPT AS FOLLOWS:

- UPON FAA-ACCEPTED REPAIR (I.E., AC 43.13-1 LATEST REVISION) OF THE WINDOW, UNPRESSURIZED FLIGHT IS PERMITTED PENDING INSTALLATION OF REPLACEMENT WINDOW;
- CRACK(S) EMANATE SOLELY FROM BOLT HOLES AND ARE WITHIN THE LIMITS PRESCRIBED FOR BOLT HOLES IN “CRACK INSPECTION - ACRYLIC WINDOWS AND WINDSHIELDS,” BELOW.

A. Perform this inspection annually or each 1,000 hours, whichever occurs first.

NOTE: This inspection should also be performed after repainting when any chemical stripping agent was used. Use the prism method described in step C, below, to inspect the entire circumference of the window for stripper damage/etching; especially in areas concealed by window frames, retainers, plates, or collars.

B. Inspect window panes as follows:

(1) Inspect sealant for signs of wear, deterioration, and positive contact with mounting surfaces.

(2) Inspect window frames, retainers, plates, and collars for cracks, loose rivets or screws, corrosion, and structural defects.

C. Inspect bolt holes in windows and co-pilot’s windshield as follows, using Window Inspection Kit - P/N 766-294. (A 45° acrylic prism and glycerin may be used if kit not available.)

(1) Clean area to be inspected with soap and water to ensure surface is free of oil, dirt, and wax.

(2) Determine which immersion oil to use: Type A in cold weather; Type B in hot weather.

(3) Apply a small amount of oil to the surface of window adjacent to bolt hole. On initial application, use a small amount of oil applied directly to one 90° face of inspection prism in contact with glass or acrylic (see “Figure 2”).

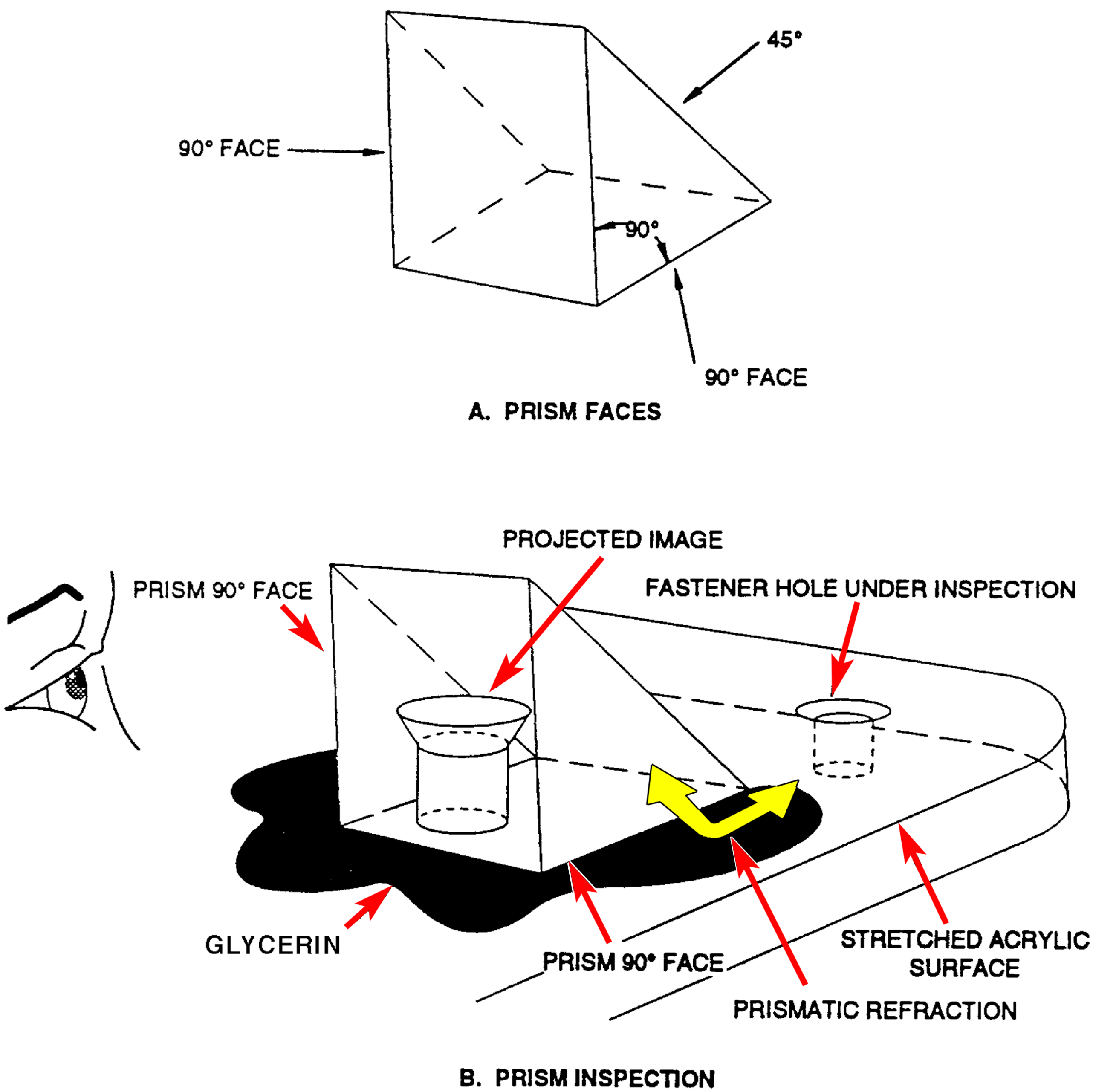
(4) Press oiled face of prism to glass or acrylic. Slide prism around until a constant film of oil extends across prism face and window surface.

(5) Look into 90° face of prism (see “Figure 2”). The image of an unfractured fastener hole will appear as a frosty cylinder.

(6) If hole is countersunk, cylinder will appear to have a cone setting on one end as seen in “Figure 2”.

(7) The image of a cracked hole will appear as a frosty or reflective projection extending from the hole (see detail A-A, “Figure 3”).

(8) The image of a crack from one hole to another will appear as a frosty or reflective irregular surface (see detail B-B, “Figure 3”).



Visual Inspection
Figure 2

D. Crack Limitation.

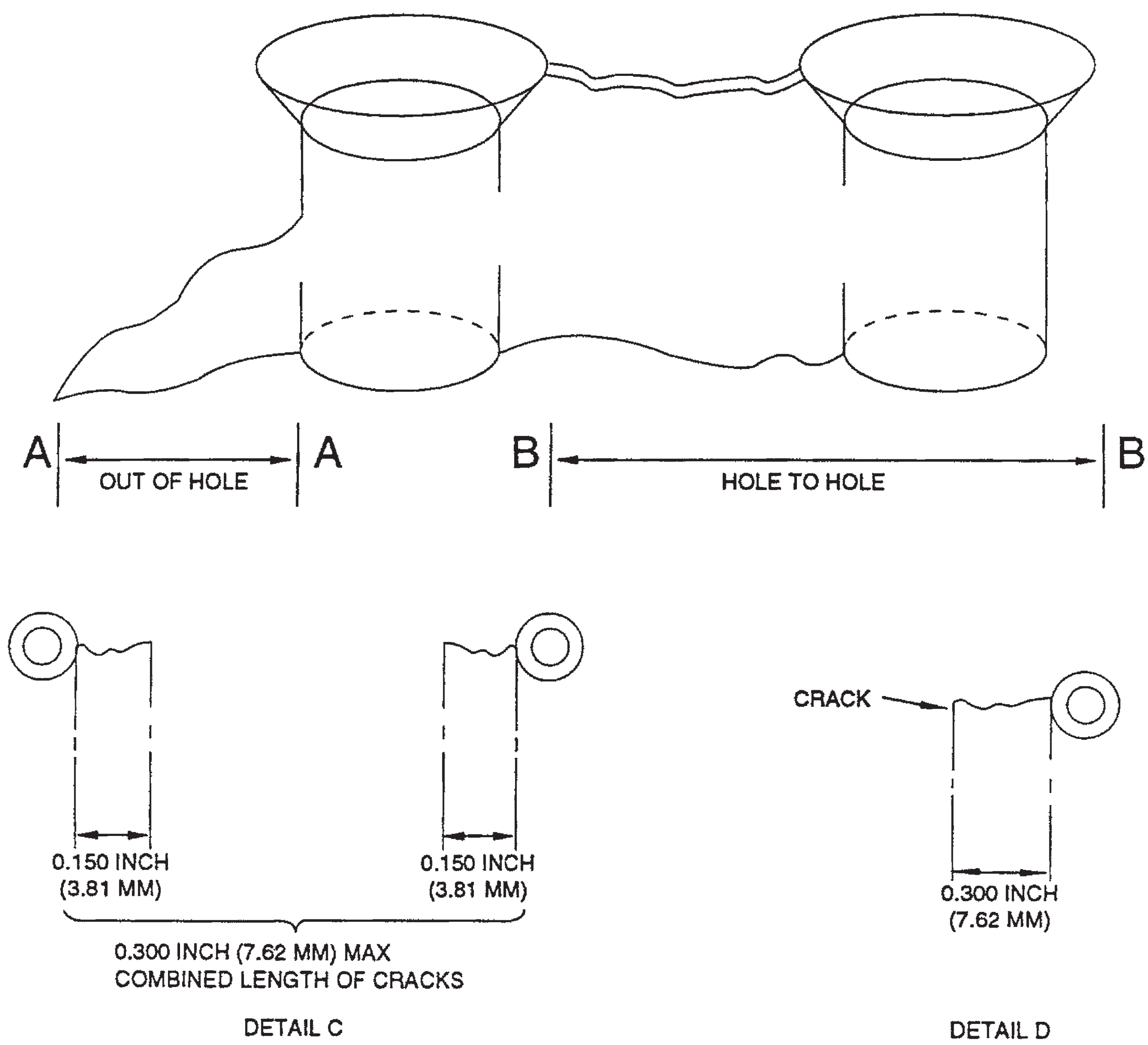
(1) The maximum acceptable length of a crack from any single bolt hole is 0.300 inch (7.62mm).

(2) The maximum combined length of multiple cracks into the space between two adjacent bolt holes is 0.300 inch (7.62mm) (see “Figure 3”).

(3) If a crack less than 0.300 inch (7.62mm) is left unrepaired, reinspect the window each additional 25 hours of aircraft operation.

(4) A maximum of three bolt holes with cracks less that 0.299 inch (7.59mm) long are acceptable in any one window pane before that window pane must be replaced.

(5) Replace window pane if a single crack or combination of adjacent facing cracks exceeds 0.300 inch (7.62mm) in length.



Crack Limitations
Figure 3

E. Chips, Scratches, and Crazing Limitations.

CAUTION: DAMAGE LIMITS AND REPAIRS STATED HEREIN MUST BE APPLIED CONSERVATIVELY. REWORK OF MULTIPLE CHIPS OR SCRATCHES SHOULD NOT BE ATTEMPTED IN ANY CASE WHERE, WHEN FINISHED, A LINEAR PATTERN WILL HAVE BEEN CREATED.

(1) Scratches in acrylic windows and the co-pilot's windshield may be reworked (see Window / Windshield Repair / Rework, below) as follows:

(a) Co-pilot's windshield - scratch is less than 0.030 inch (0.762 mm) deep and windshield is a minimum of 0.279 inch (7.087 mm) thick after rework.

(b) Cockpit side windows - scratch is less than 0.025 inch (0.635 mm) deep and window is a minimum of 0.225 inch (5.715 mm) thick after rework.

(c) Cabin side windows - scratch is less than 0.021 inch (0.533 mm) deep and window is a minimum of 0.189 inch (4.800 mm) thick after rework.

(d) No vision distortions in critical and semi-critical areas as a result of rework.

(2) Replace acrylic windows with scratches that cannot be reworked in accordance with these standards.

6. Water Leak Check

After any window/windshield repair/rework, perform a Water Leak Check, 20-00-00.

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FLIGHT COMPARTMENT

1. Windshield

A. Removal

(1) If removing the pilot's windshield (i.e. - heated), disconnect the battery and remove external power connections, if any.

(2) Remove all screws which attach the windshield to the retainer and the retainer to the fuselage.

(3) If removing heated windshield, disconnect black ground wire from tooling hole near instrument panel support clip.

(4) Remove the retainer.

(5) Remove the windshield.

B. Installation

(1) If the old retainer is being reused, remove all traces of the old sealant from its inside surface.

(2) Remove all traces of the old sealant from the shims, windshield mounting angles and retainer mating surfaces.

(3) All corners must be broken and all edges smooth and free of tool marks.

(4) Check fit the windshield to the fuselage to ensure that it will fit the contour of the fuselage. If replacing the co-pilot's windshield or non-heated pilot's windshield (i.e. - acrylic), trim excess material as required.

(5) If using a new retainer:

(a) Check fit and trim to fit as required. The areas most likely to require trimming are the lower left outboard side and along the bottom edge near the center post.

(b) Using the old retainer as a template, drill .191 inch screw holes. Countersink the screw holes .380 in. dia. x 100 degrees.

NOTE: For the pilot's heated glass windshield, apply a parting agent (Polyvinyl Alcohol, P/N 915-936, or equivalent) to the faying surface of the retainer and do not install silicone rubber tubes in the screw holes as described in step (6), below.

(6) Apply sealant per Figure 1 and install windshield ensuring that a silicone rubber tube is installed in each tooling hole in the windshield. Torque any fasteners which penetrate the windshield as specified in Figure 1.

(7) If installing glass heated windshield, connect black ground wire to tooling hole near instrument panel support clip with a MS35206-215 screw, AN960-4 washer, MS35338-41 lock washer, and a MS21044N04 nut.

(8) Install the retainer.

(9) Conduct the Pressurization System Test, 21-30-00, as required.

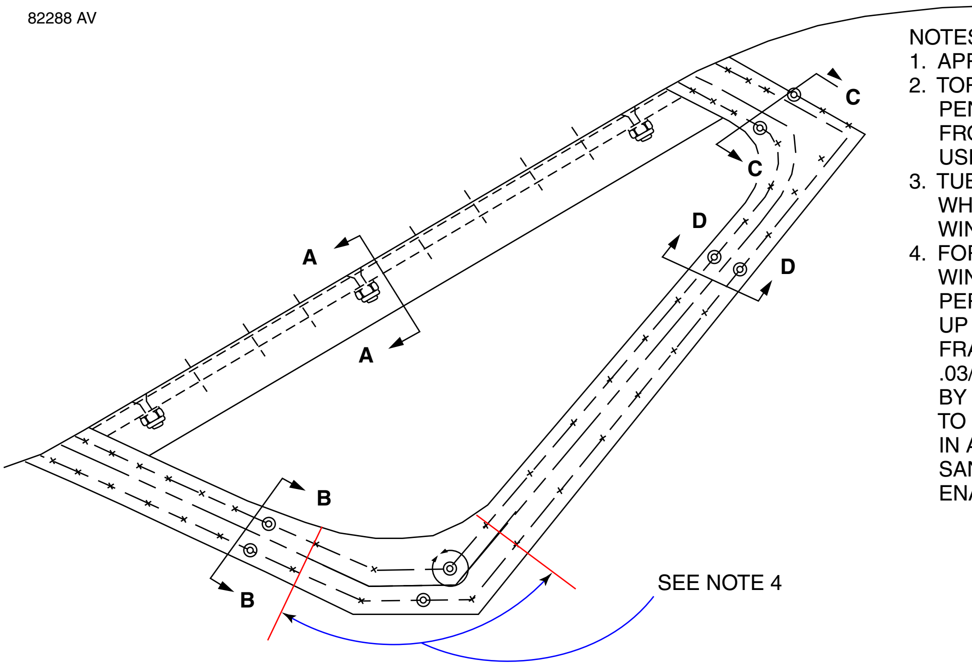
(10) Reconnect the battery and, if desired, external power.

(11) Conduct the Heated Windshield Test, 30-40-00, as required.

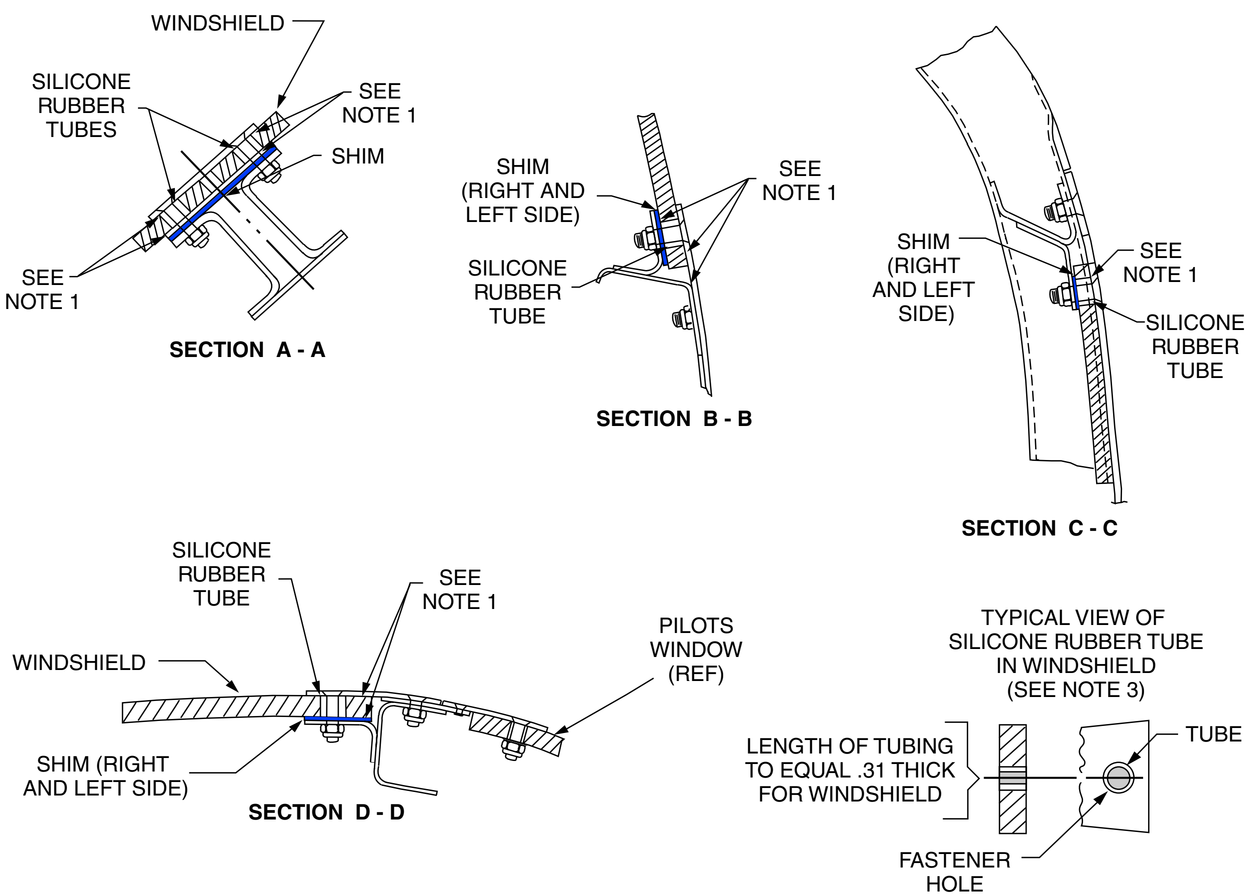
(12) Perform a Water Leak Check, 20-00-00.

(13) Prime and paint as desired.

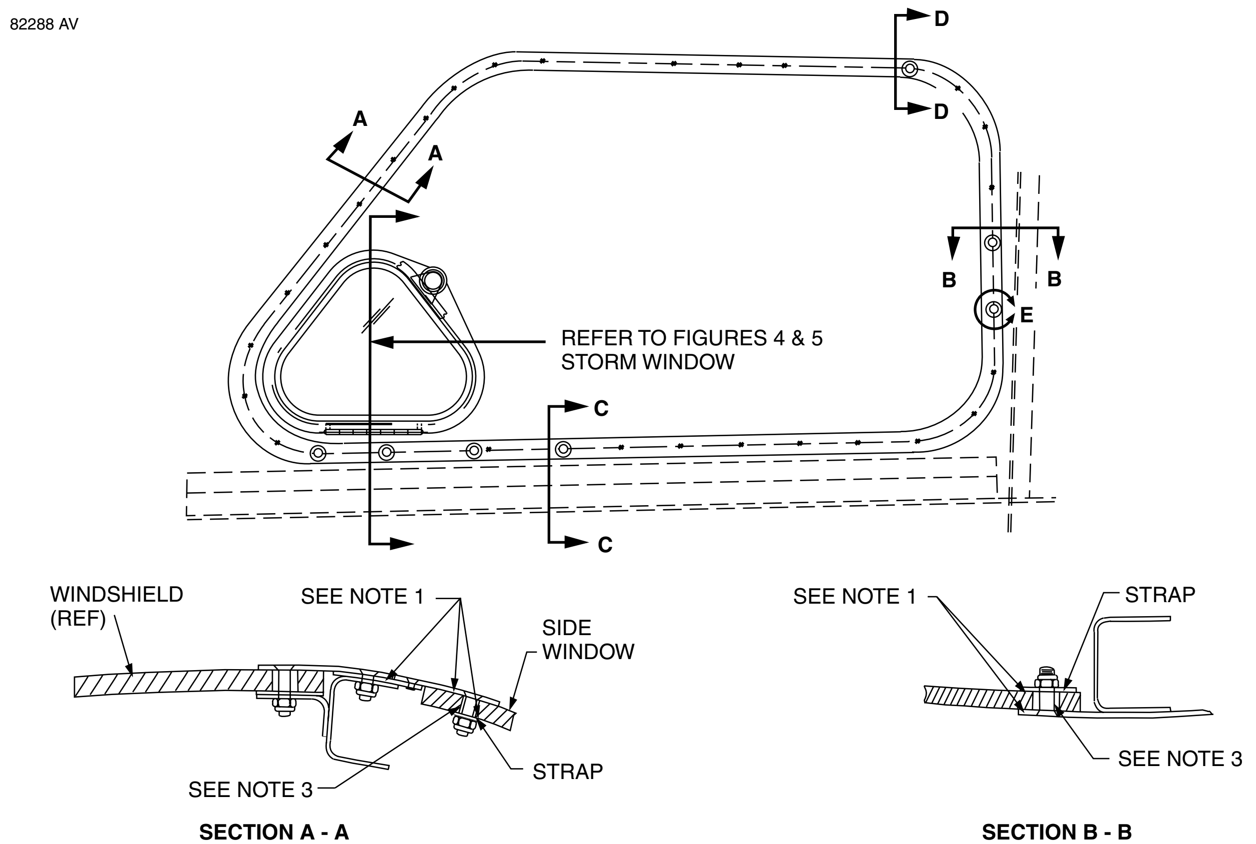
(14) Make a proper logbook entry documenting this installation.



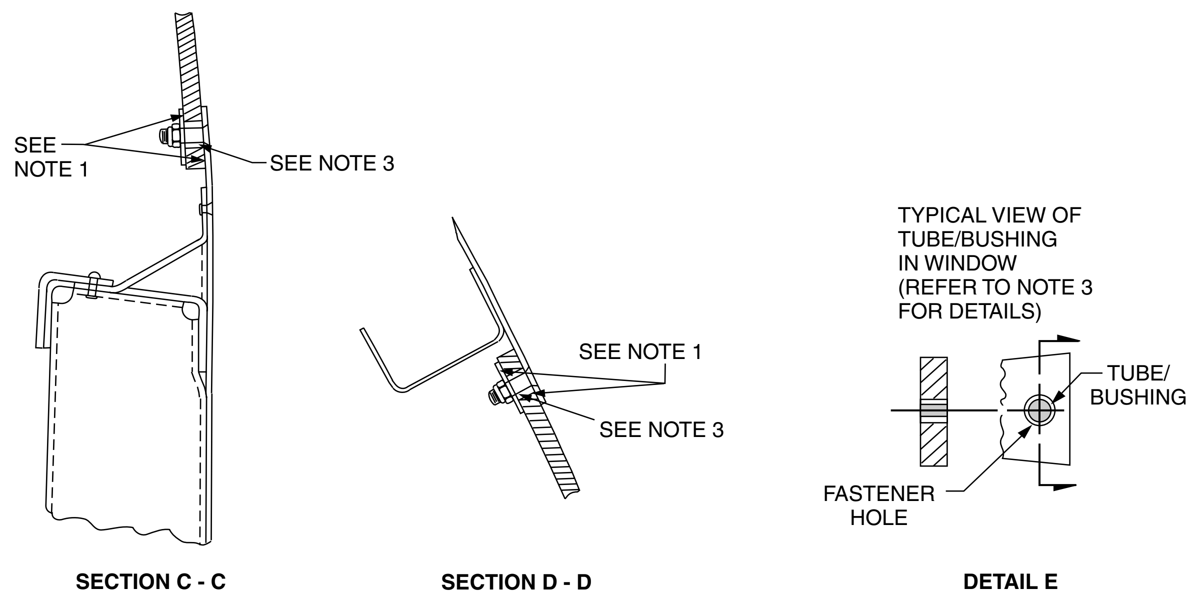
NOTES:
1. APPLY SEALANT HERE.
2. TORQUE ALL FASTENERS WHICH PENETRATE THE WINDSHIELD FROM 19 TO 22 INCH POUNDS USING A STAR PATTERN.
3. TUBING NOT REQUIRED WHEN GLASS HEATED WINDSHIELD IS INSTALLED.
4. FOR PILOT'S HEATED GLASS WINDSHIELD ONLY, IT IS PERMISSIBLE TO REMOVE UP TO .10 INCH OF FIBERGLASS FRAME EDGE TO ACHIEVE .03/.06 INCH EDGE CLEARANCE BY SANDING PERPENDICULAR TO EDGE WITH (MAX) 220 GRIT, IN AREA SHOWN ONLY. REFINISH SANDED EDGE WITH FLAT BLACK ENAMEL PAINT.



Windshield Installation
Figure 1



NOTES:
1. APPLY SEALANT HERE. SEE 91-10-00, CHART 10, SEALANT, WINDOWS AND WINDSHIELDS.
2. ALL FASTENERS WHICH PENETRATE THE WINDOW SHALL BE TORQUED TO 19 - 22 INCH POUNDS IN A STAR PATTERN.
3. PA-46-350P: 0.25 LONG SILICONE RUBBER TUBE, PA-46R-350T: 0.32 LONG ALUMINUM BUSHING.



Pilot's / Copilot's Window Installation
Figure 2

2. Pilot's / Copilot's Window

NOTE: In S/N's 4636425 and up and S/N's 4692001 and up, the pilot's side window is a one piece installation and does not include a storm window.

A. Removal

(1) Remove the screws, nuts, washers and strap that retain the window to the fuselage.

(2) Remove the window.

B. Installation

NOTE: In S/N's 4636001 thru 4636076 only, if installing a service replacement pilot's side window, order Kit No. 767-355.

(1) Remove all traces of old sealant from the outboard side of the strap and the inboard side of the fuselage window frame.

(2) All corners must be broken and all edges smooth and free of tool marks.

(3) Check fit the window to the fuselage window frame to ensure that it will fit. Trim excess material as required.

NOTE: In S/N's 4692001 and up, when installing a side window, apply two separate coats of Partall Paste #2 paste wax to the faying surfaces, approximately 2 inches around the outer edge of window, on the inboard side only. Apply with a clean SAE AMS3819 Class 1 grade A cleaning cloth. After each coat, allow paste to dry for approximately 2 minutes and buff with a clean SAE AMS3819 Class 1 grade A cleaning cloth. A thin layer of paste wax should remain on window upon completion.

(4) Apply sealant to the inboard side of the fuselage window frame.

(5) Position the window in its frame, apply sealant to the outboard side of the strap. Secure the window to the frame and strap with the previously removed nuts, bolts and washers. Torque any fasteners which penetrate the window as specified in Figure 2.

(6) Perform a Water Leak Check, 20-00-00.

C. 50 Hour Pilot's Side Cockpit Window/Storm Window Inspection - PA-46-350P only

NOTE: This inspection incorporates the requirements of Piper Service Bulletin No. 1175B.

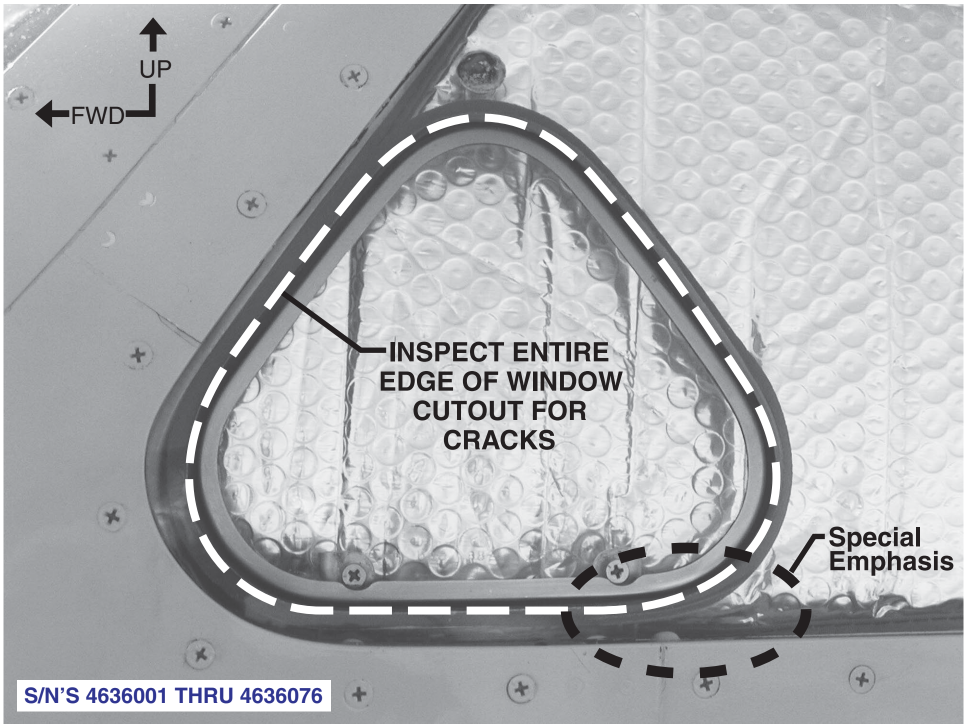
NOTE: Installation of replacement window P/N's 82282-013 or 82282-022 (S/N's 4636001 thru 4636221) or P/N's 82282-014 or 82282-020 (S/N's 4636222 thru 4636424), relieves the requirement for this inspection.

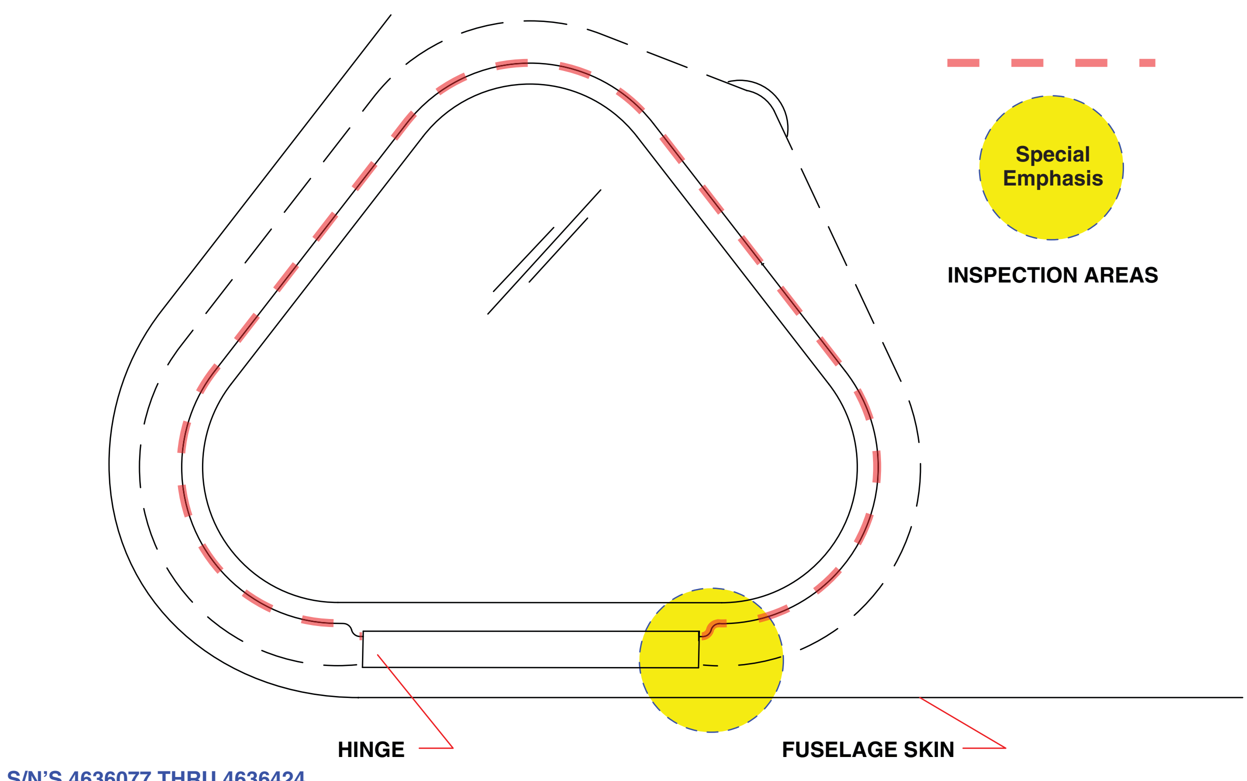
NOTE: Production of storm windows has been discontinued. Once the remaining stock is consumed, pilot's side windows with storm windows that fail inspection or are otherwise damaged must be replaced by the new style window.

In PA-46-350P S/N's 4636001 through 4636424 only, for airplanes with pilot's side cockpit windows that are equipped with a storm window and that side cockpit window has accumulated 350 hours time-in-service, upon reaching 350 hours and each 50 hours time-in-service thereafter, inspect as follows:

Carefully examine the edge of the entire storm window cutout for the existence of cracks with special emphasis on the areas circled in "Figure 3". "Figure 4" provides examples of typical cracks, which initiate at the edge of the cutout, and can grow in length, reducing the strength of the window.

(1) If no crack is found, make an appropriate logbook entry documenting completion of this inspection and return airplane to service. Repeat this inspection at 50 hour intervals until a replacement side cockpit window without a storm window is installed.





LOOKING INBOARD FROM LEFT

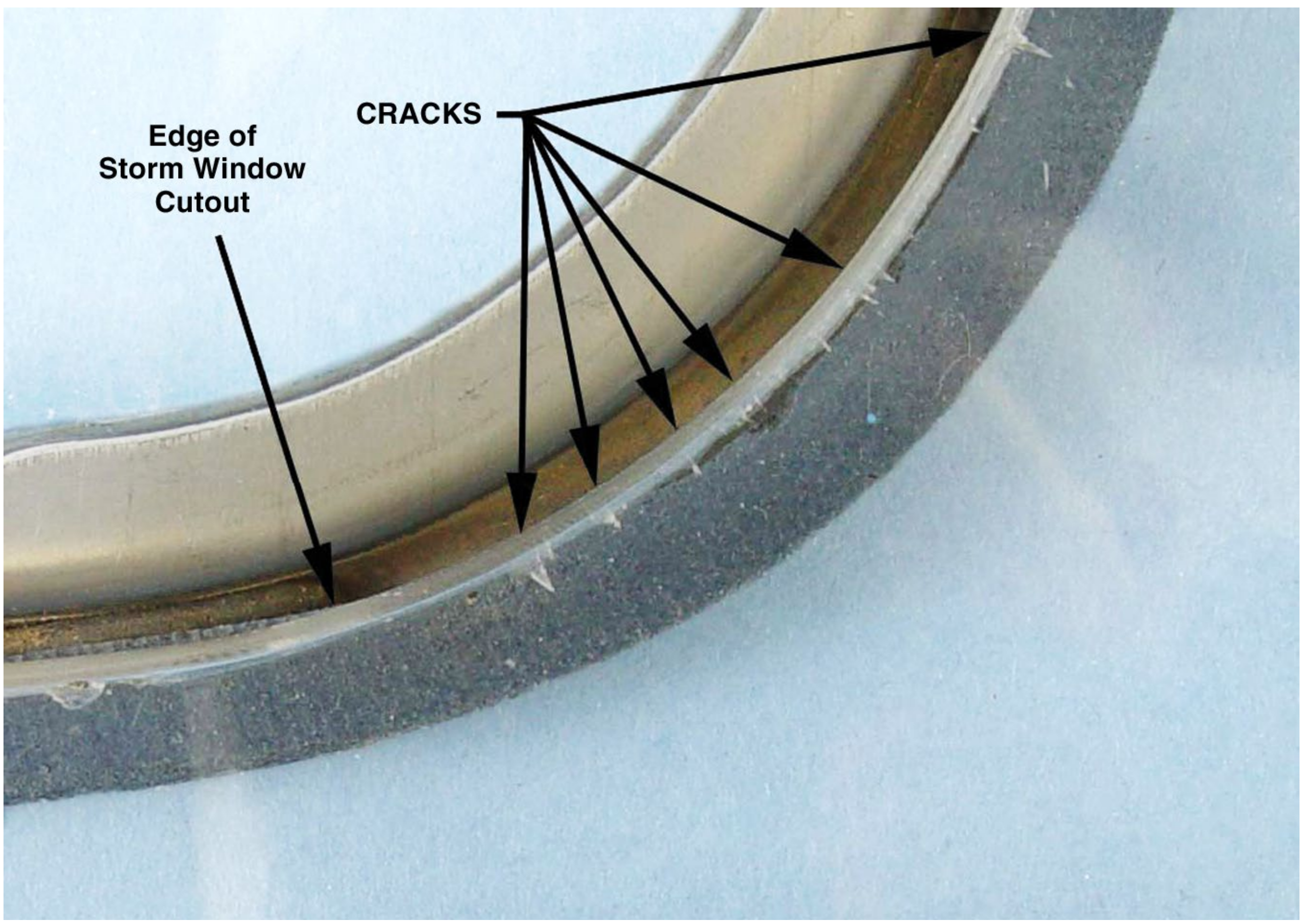
Pilot's Side Cockpit Window Inspection
Figure 3

Effectivity
PA-46-350P
4636001 thru 4636424

CAUTION: PRESSURIZED FLIGHT PROHIBITED WITH CRACKED WINDOW.

(2) If any crack is found, pressurized flight is prohibited until the cracked side cockpit window is replaced per installation, above.

NOTE: Unpressurized flight operations are authorized pending replacement of a cracked window.



Effectivity
PA-46-350P
4636001 thru 4636424

Cracks at the Storm Window Cutout
Figure 4

3. Storm Window (S/N's 4636001 thru 4636076 only)

See "Figure 5".

A. Removal

(1) Rotate the latch to open the storm window.

(2) Remove the nuts, washers, bolts and screws that attach the storm window hinge to the pilot's window.

(3) Remove the storm window.

B. Installation

(1) Position the storm window in the hole in the pilot's window.

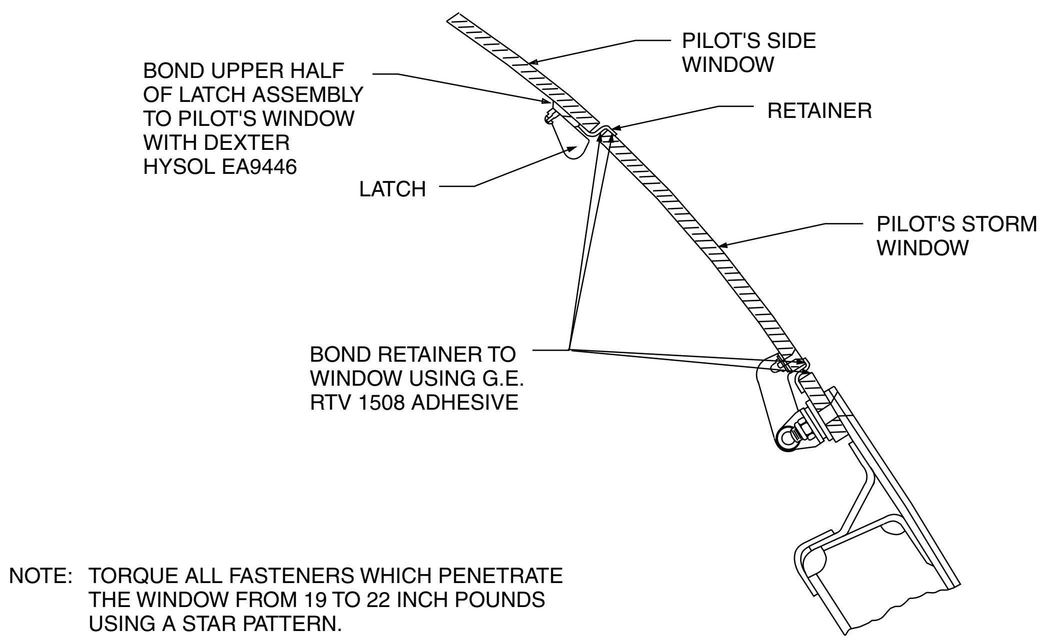
(2) Torque the fasteners that attach the storm window hinge to the window as specified in Figure 4.

(3) Rotate the latch to lock the storm window.

NOTE: The storm window retainer is bonded to the storm window with General Electric RTV1508 adhesive or equivalent (i.e. - G.E. RTV-108, DOW RTV-732, etc.). The latch assembly upper half is bonded to the pilot's window with Dexter Hysol EA9446.

(4) Perform a Water Leak Check, 20-00-00.

NOTE: If storm window fails "Water Leak Check", Add up to 0.15 RTV-732 (P/N 279-195) between seal and window retainer and perform "Water Leak Check" again.



Storm Window
Figure 5

Effectivity
PA-46-350P
4636001 thru 4636076

4. Storm Window (S/N's 4636077 thru 4636424 only)

A. Replacing the storm window without replacing the retainer assembly. (See "Figure 6".)

(1) Removal

(a) Push button on latch and pull latch to open the storm window.

(b) Remove hinge pin.

(c) Remove storm window.

(2) Installation

(a) Align storm window hinge half with hinge half on the pilot's window.

(b) Install hinge pin.

(c) Push latch onto latch pin to close the storm window.

(d) Perform a Water Leak Check, 20-00-00.

NOTE: If storm window fails "Water Leak Check", Add up to 0.15 RTV-732 (P/N 279-195) between seal and window retainer and perform "Water Leak Check" again.

B. Installing a storm window and retainer assembly in a new pilot's side window.

(1) Position the hinge in the notch at the base of the storm window cut-out. With the storm window in a closed position, center the storm window in the cut-out and mark the pilot's window to indicate the outer edges of the hinge.

(2) Bond the hinge half to the pilot's side window with Dexter Hysol EA9446 (see A-A, "Figure 6"). Take care to ensure the inside edge of the hinge half is bonded flush to the pilot's window. Also, ensure that a .030 inch gap is left between the bottom edge of the cut-out and the inside of the hinge half and that the gap is filled with adhesive.

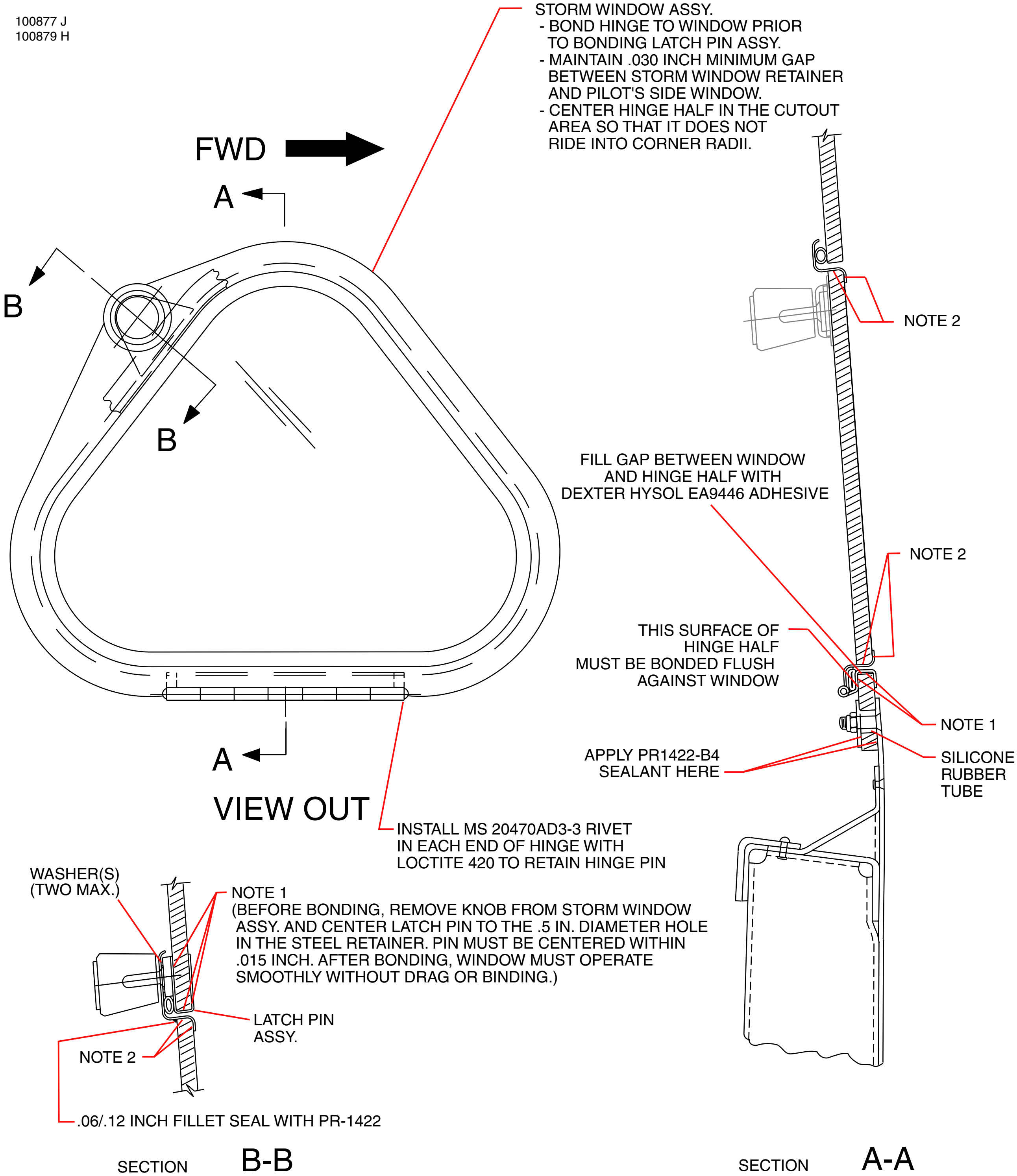
(3) When the hinge to window bond is cured, slide the latch pin assembly over the rear upper edge of the storm window cut-out and push the storm window to a closed and latched position. Mark the position of the latch pin assembly on the pilot's side window.

(4) Bond the latch pin assembly to the pilot's side window with Dexter Hysol EA9446 (see B-B, "Figure 6"). Ensure that the latch pin assembly is aligned to the markings previously made on the pilot's side window.

NOTE: The storm window retainer is bonded to the storm window with PRC PR-1422 Sealant. The latch pin assembly and hinge are bonded to the pilot's window with Dexter Hysol EA9446.

(5) Perform a Water Leak Check, 20-00-00.

NOTE: If storm window fails "Water Leak Check", Add up to 0.15 RTV-732 (P/N 279-195) between seal and window retainer and perform "Water Leak Check" again.



NOTES: 1. BOND WITH DEXTER HYSOL EA9446.
2. BOND WITH PR-1422.
3. IF STORM WINDOW FAILS "WATER LEAK CHECK," ADD UP TO 0.15 RTV-732 (P/N 279-195) BETWEEN SEAL AND WINDOW RETAINER AND PERFORM "WATER LEAK CHECK" AGAIN.

Effectivity
PA-46-350P
4636077 thru 4636424

Storm Window
Figure 6

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CABIN

Passenger Windows

A. Removal

(1) Drill out the rivets which fasten the window retainer to the fuselage skin.

(2) Push the window and retainer inward until it is free of the fuselage skin.

(3) Remove the retainer from the window.

(4) Save the old window for a template if a new window is to be installed.

B. Installation

(1) Remove all traces of old sealant from the retainer, inboard edge of the window opening, and old window.

(2) All corners must be broken and all edges smooth and free of tool marks.

(3) If a new window is being installed, ensure that it is the same dimensions as the old window.

(4) Sealant

(PIR-PPS45012-1, Rev. AL., PMS-C1012, Rev. J.)

(a) On PA-46-350P aircraft, apply PRC 1425-B2 sealant to the outboard surface of the window where it mates with the inboard edge of the fuselage skin window opening. Also apply sealant to the outboard surface of the retainer where it mates with the inboard edge of the fuselage skin window opening and where it mates with the inboard edge of the window.

(b) On PA-46R-350T aircraft, fillet seal where outboard surface of the window mates with the inboard edge of the fuselage skin window opening with Bostik 1100 FS (Urethane).

(5) Line up the rivet holes and rivet the retainer to the skin.

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DOOR

Door Window

A. Removal

(1) Drill out the rivets which fasten the window retainer to the inner frame of the door.

(2) Remove the retainer.

(3) Remove the door window.

B. Installation

(1) Remove all remaining traces of a sealant from the inner surface of the door window frame and outboard-facing side of the retainer.

(2) Fit-check the new window before applying sealant.

(3) Sealant

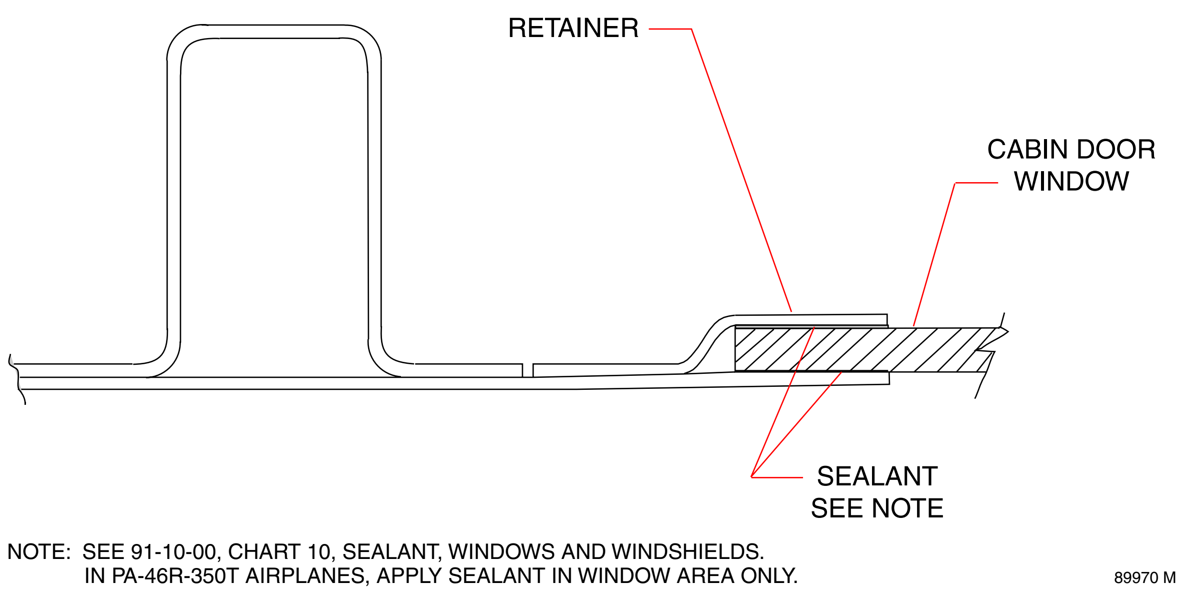
(PIR-PPS45012-1, Rev. AL., PMS-C1012, Rev. J.)

(a) On PA-46-350P aircraft, apply PRC 1425-B2 sealant to the outboard surface of the window where it mates with the inboard edge of the door frame window opening. Also apply sealant to the outboard surface of the retainer where it mates with the inboard edge of the door frame window opening and where it mates with the inboard edge of the window.

(b) On PA-46R-350T aircraft, fillet seal where outboard surface of the window mates with the inboard edge of the door frame window opening with Bostik 1100 FS (Urethane).

NOTE: Apply sealant in window area only.

(4) Line up the rivet holes and rivet the retainer to the frame.



Section View of Cabin Door Window
Figure 1

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CHAPTER

57

WINGS

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CHAPTER 57

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
57-List of Effective Pages	1	Mar 25/21			
	2	Sep 15/09			
57-Table of Contents	1	Mar 25/21			
	2	Sep 15/09			
57-40-00	1	Dec 15/14			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Sep 15/09			
	5	Dec 15/14			
	6	Sep 15/09			
	7	Dec 15/14			
	8	Sep 15/09			
	9	Sep 15/09			
	10	Sep 15/09			
	11	Sep 15/09			
	12	Sep 15/09			
	13	Mar 25/21			
	14	Sep 15/09			
	15	Mar 25/21			
	16	Sep 15/09			
	17	Nov 30/17			
	18	Sep 15/09			
	19	Mar 25/21			
	20	Mar 25/21			
	21	Mar 25/21			
	22	Mar 25/21			
57-50-00	1	Sep 15/09			
	2	May 31/11			
	3	May 31/11			
	4	Dec 15/14			

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CHAPTER 57 - WINGS

TABLE OF CONTENTS

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ATTACH FITTINGS

1. General

See ”Figure 1”.

This chapter contains removal and installation procedures for the wing and related components.

The wing is comprised of three separate sections. The center section is comprised of the inboard portion of each wing joined by a continuous main spar. The outboard edges of each inboard portion of the wing are fitted with an outer panel splice. The outboard panel has a mating splice plate and an integral fuel tank from its center splice to an area 18 inches from the wing tip.

NOTE: Replacement wings and wing sections built for service are direct replacement parts for current production aircraft only (S/N’s 4636196 and up and S/N’s 4692001 and up). When replacing an outboard wing section in these airplanes, order Piper Instructions and Hardware Service Kit No. 88398-002 (Left) or 88398-003 (Right) in addition to the appropriate outboard wing assembly.

S/N’s 4636001 thru 4636195 must order Piper Instructions and Hardware Service Kits as listed below to enable installation of a new replacement wing or wing section(s):

To Install a New	Order Hardware Kit No.
Left Outboard Wing Section	766-662
Right Outboard Wing Section	766-663
Center (Inboard) Wing Section	766-665
Complete Wing	766-675

NOTE: When replacing the inboard wing section, either or both outboard wing section, and/or the entire wing, have your Piper Dealer order the base service wing part number 106896-002.

Additionally, you must specifiy “complete wing” or which wing section is required as follows:

“LH Outboard,” “RH Outboard,” and/or “Center.”

2. Outboard Wing

A. Removal

See ”Figure 1” and “Figure 2”.

(1) Drain the fuel from both wings (refer to Draining Fuel System, 12-10-00).

(2) Remove the aileron per 57-50-00. If replacing the wing section, remove the aileron sector.

(3) Remove the flap per 57-50-00.

(4) Carefully remove the wing splice skin (W.S. 107.56 - W.S. 115.25) that covers the inboard / outboard wing splice. This old skin may be used as a template for the new skin.

(5) Remove the screws from the fuel access panels.

NOTE: The fuel access panels are installed with low adhesive sealant and will not come off without some method of separating them from the wing.

(6) Using a fabricated laminated fiberglass wedge, carefully remove the fuel access panels for reuse.

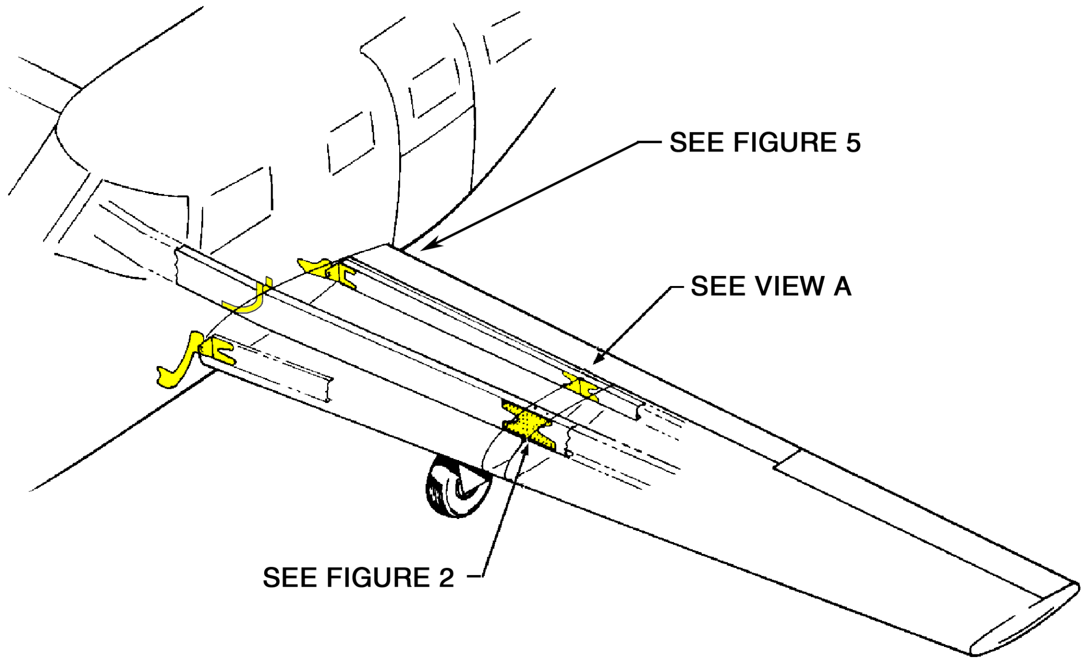
(7) Disconnect the fuel lines by removing clamps and cutting the flexible connections.

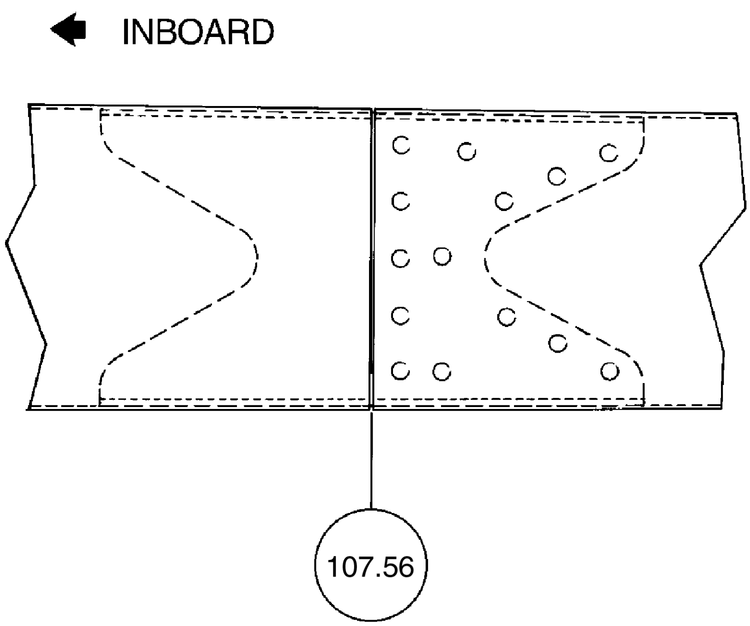
(8) Disconnect the fuel quantity sender lead. If replacing the wing section, remove fuel quantity sender per 28-40-00.

(9) Disconnect the taxi/recognition and navigation/strobe light leads. If replacing the wing section, remove the taxi/recognition and navigation/strobe lights per 33-40-00.

(10) Support outboard wing. Drill out rivets and remove Hi-Lok pins and collars from the splice plate.

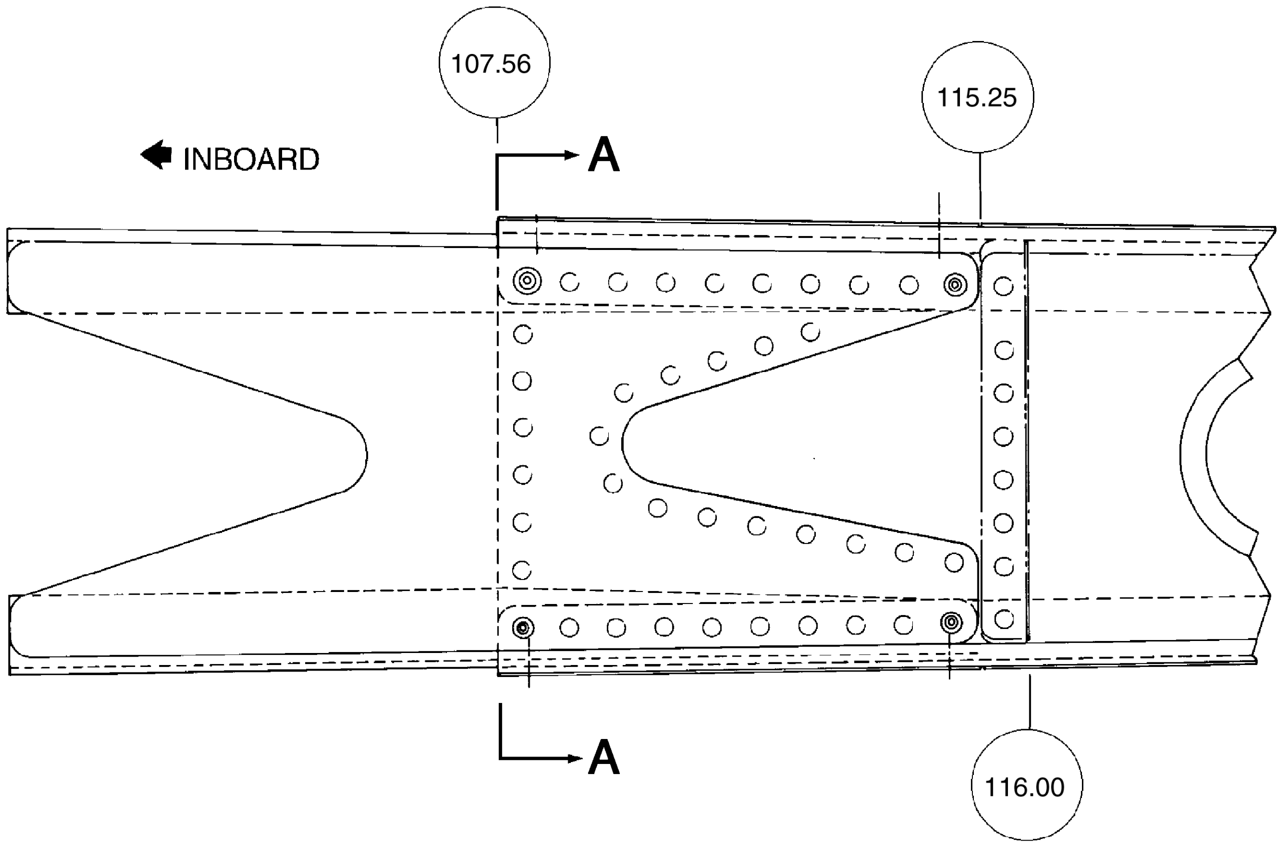
(11) Carefully remove the outboard wing, being certain that all electrical leads, fuel and vent lines, and control cables are disconnected.

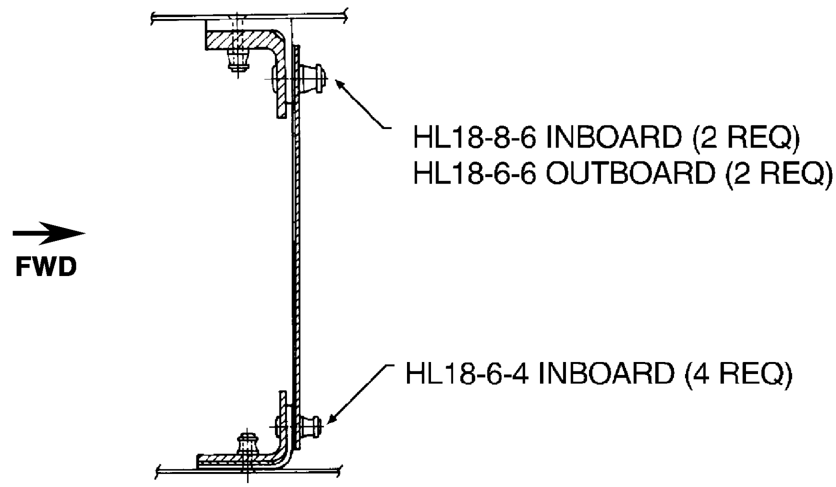




VIEW A
(LOOKING AFT)

Wing Installation
Figure 1

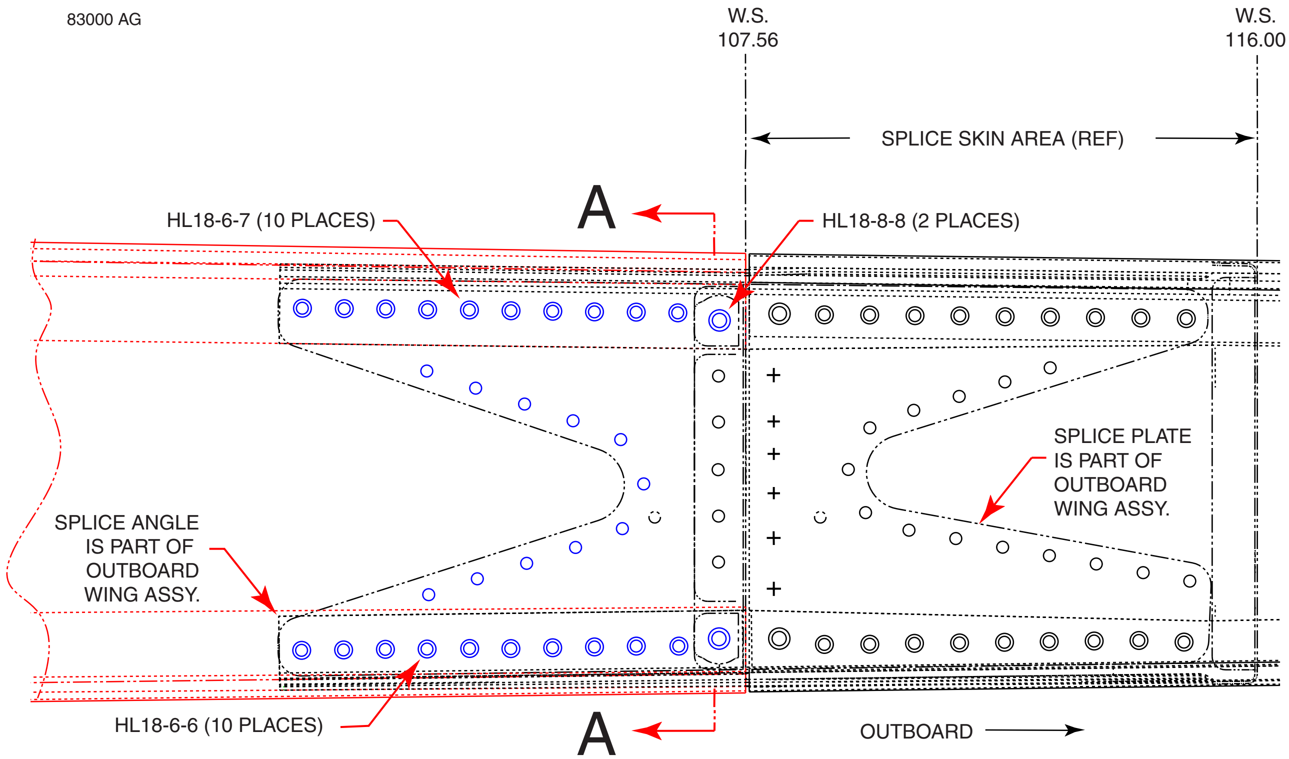


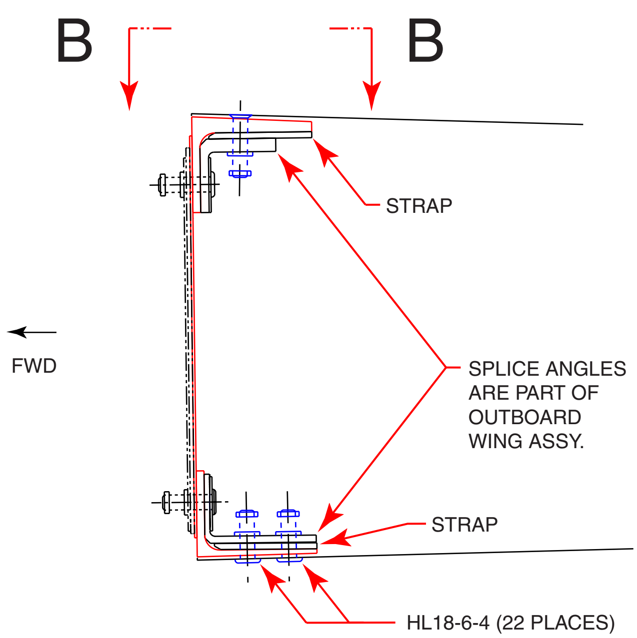


VIEW A - A
(LOOKING OUTBOARD)

Outboard Wing Installation
Figure 2 (Sheet 1 of 3)

Effectivity
4636001 thru 4636195

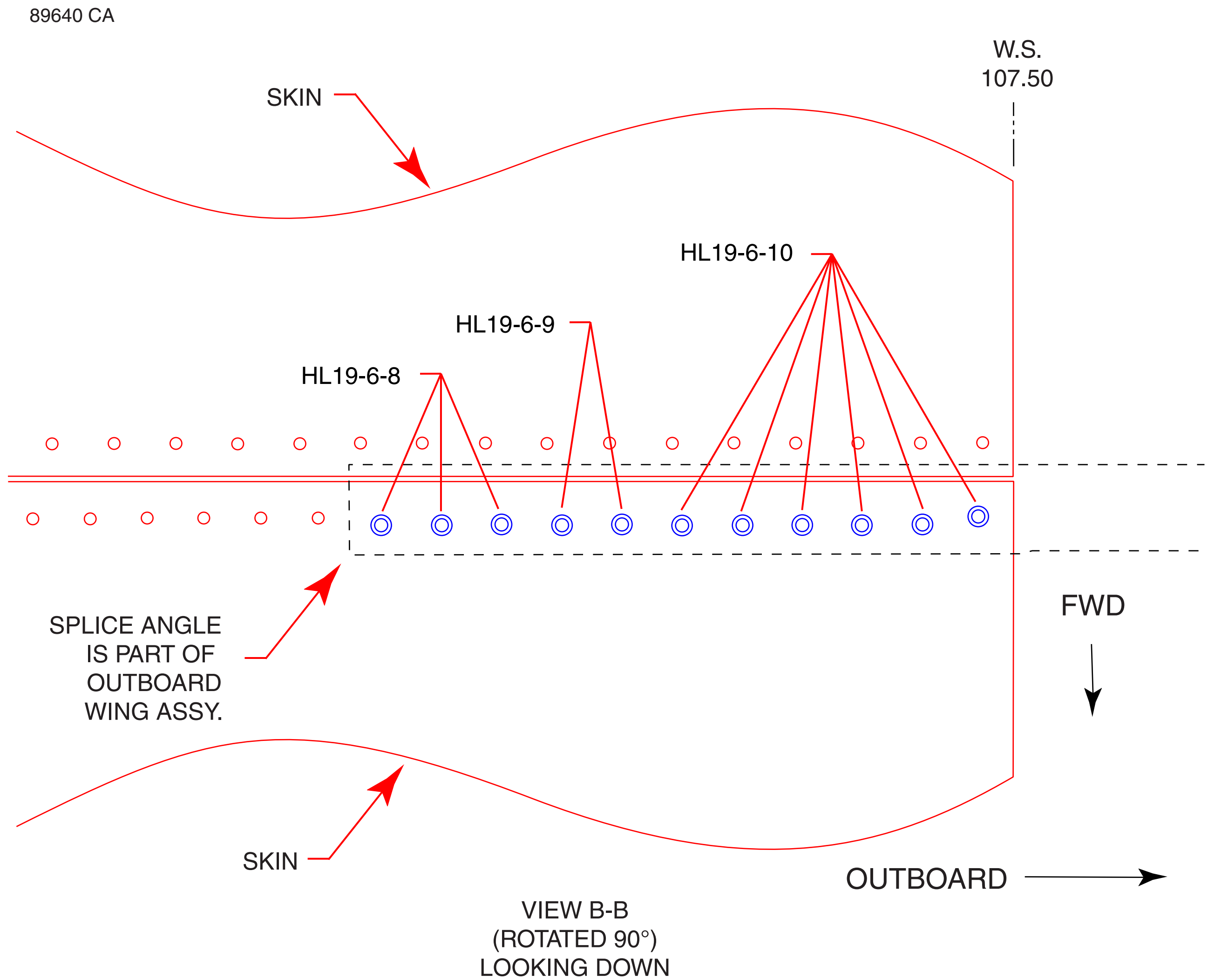




VIEW A-A
(LOOKING INBOARD)

Effectivity
4636196 & up
4692001 & up

Outboard Wing Installation
Figure 2 (Sheet 2 of 3)



Outboard Wing Installation
Figure 2 (Sheet 3 of 3)

Effectivity
4636196 & up
4692001 & up

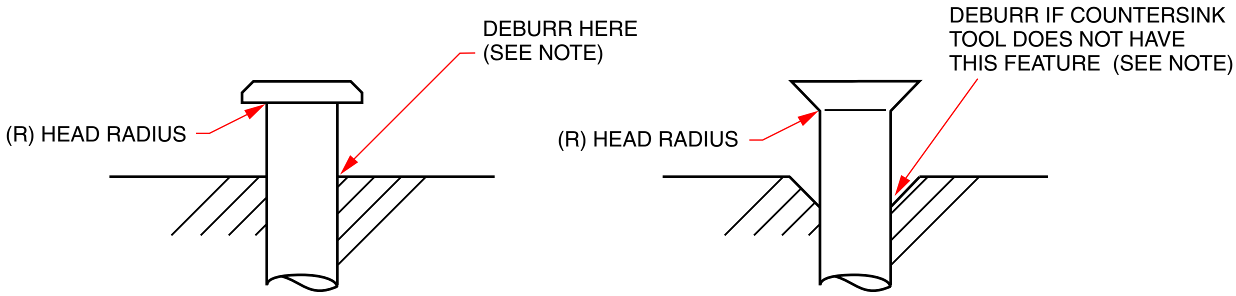
B. Installation

NOTE: When installing a "replacement" wing or wing section, perform Stall Warning Flight Test Procedure, 27-30-00, upon completion of wing installation.

(1) Carefully position the outboard wing in alignment with the splice plates on the inboard wing main and rear spars.

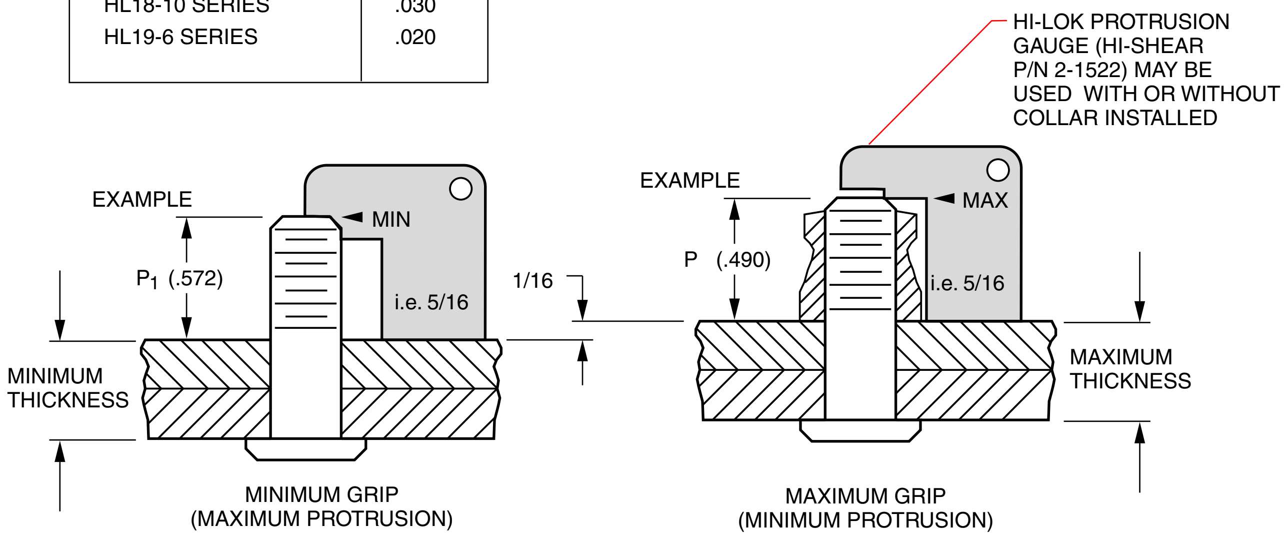
(2) Using a string of suitable length extended from the inboard wing leading edge next to the fuselage out to the outboard wing tip to help maintain true alignment of the wing sections. Also, extend a string from the wing tip to the stabilizer, marking its length and checking this with the opposite wing and stabilizer, using the same reference points on both wings and stabilizers.

(3) While maintaining proper alignment of the wing, install the Hi-Lok pins and rivets into the spars and splice plates per "Figure 3". Refer to the appropriate Piper Illustrated Parts Catalog for correct rivet part numbers and locations.



NOTE: The HI-LOK pin has a slight radius under its head. After drilling, deburr the edge of the hole. This permits the head to fully seat in the hole. See the list in this figure for head radius dimensions. For example, the 3/16 protruding head (HL 18-6 series) has a .015/.025 radius while the 3/16 flush head (HL 19-6 series) has a .025/.030 radius.

TYPE OF HI-LOK PIN	(R)
HL18-5 SERIES	.025
HL18-6 SERIES	.015
HL18-8 SERIES	
HL18-10 SERIES	.030
HL19-6 SERIES	.020



EXAMPLE
HL18-10-10

MAXIMUM GRIP LENGTH OF PIN IN 1/16ths
(10/16" OR 5/8" GRIP LENGTH FOR THIS EXAMPLE)

STANDARD HI-LOK PIN FIRST DASH NUMBER	NOMINAL DIAMETER	MINIMUM PROTRUSION P	MAXIMUM PROTRUSION P_1
-5	5/32	.302	.384
-6	3/16	.315	.397
-8	1/4	.385	.467
-10	5/16	.490	.572

HI-LOK Fasteners Installation
Figure 3

57406

(4) Connect the leads from the taxi/recognition and navigation/strobe lights. If installing a replacement wing section, install the taxi/recognition and navigation/strobe lights first per 33-40-00.

(5) Connect the fuel and vent lines, using new flexible connections.

(6) Connect the fuel quantity sender leads. If installing a replacement wing section, install the fuel quantity sender first per 28-40-00.

NOTE: When installing the fuel area access panels with the sender units mounted, strict attention should be payed to avoid having the sender unit wires from becoming wrapped around the sender unit arm and immobilizing the sender when it is finally installed.

(7) Install the wing splice skin at W.S. 107.56 - 115.25.

(8) Install the flap and aileron per 57-50-00. If installing a replacement wing section, install the aileron drive sector on the aft spar first. Torque sector mounting bolt per Figure 4, 27-10-00.

(9) Check fuel system for leaks and flow (see 28-10-00).

(10) Check operation of the taxi/recognition lights and navigation/strobe lights.

(11) Fill the fuel system per Filling Fuel Cells, 12-10-00.

(12) Make appropriate logbook entry.

3. HI-LOK Removal and Installation

NOTE: Refer to Hi Shear Catalogs for further information and a complete list of power and hand tools to aid in removal and installation of the Hi-Lok fasteners.

A. Hi-Lok pins and collars are used at various points in the wing installation. Refer to the appropriate Piper Illustrated Parts Catalog for locations and proper pin and collar part numbers.

B. To remove the Hi-Lok pin and collar:

(1) Grasp the collar with pliers.

(2) Insert an Allen hex wrench in the hex in the end of the pin.

(3) Turn the collar with pliers.

(4) Remove the collar and pin.

(5) If not damaged during removal, the Hi-Lok pin can be reused.

C. To install the Hi-Lok pin and collar:

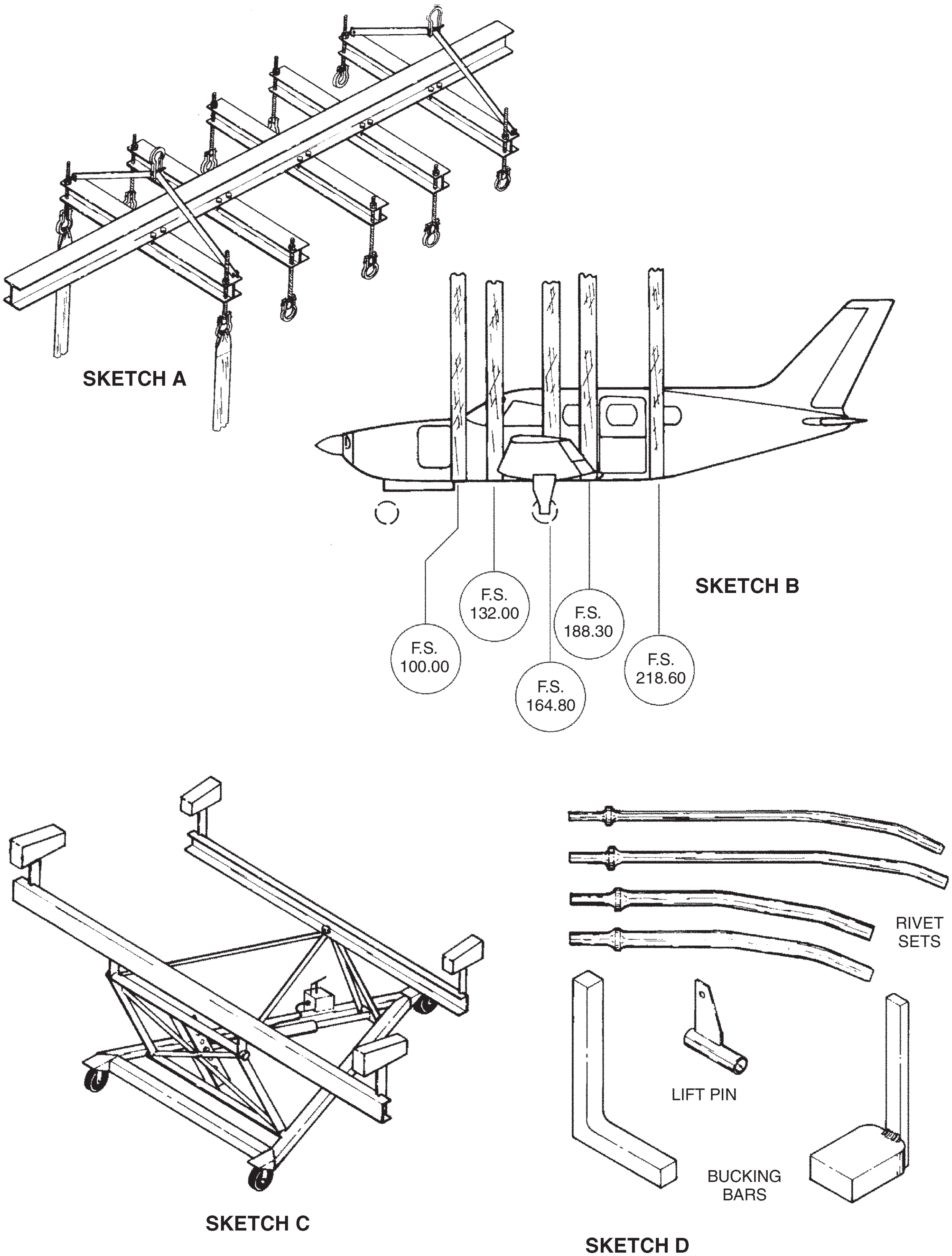
(1) Insert the Hi-Lok pin in the hole (after hole has been deburred if new).

(2) Manually turn the Hi-Lok collar onto the pin.

(3) If using Hi-Lok power tools, insert the hex wrench tip of the power driver into the pin's hex recess, and the socket over the collar hex. Press the power driver against the collar and operate the power driver until the collar's wrenching device has been torque off.

(4) If using shop tools, insert an Allen wrench into the pin's hex recess and turn the collar hex with an open end wrench until the collar's wrenching device has been turned off.

(5) Check the protrusion of the threaded end for the limits given in “Figure 3”. Hi-Lok Protrusion Gauges (P/N 2-1522) offer a convenient method to check the pin protrusion limits.



Special Equipment for Wing Removal and Installation
Figure 4

4. Inboard Wing

A. General

If it is necessary to replace the entire wing structure, you will need special equipment. Allow time for setting up as follows:

(1) Determine whether wing will be lowered or fuselage raised to remove wing. Equipment will vary accordingly.

(2) Fabricate the following special equipment prior to starting the procedure (see “Figure 4”).

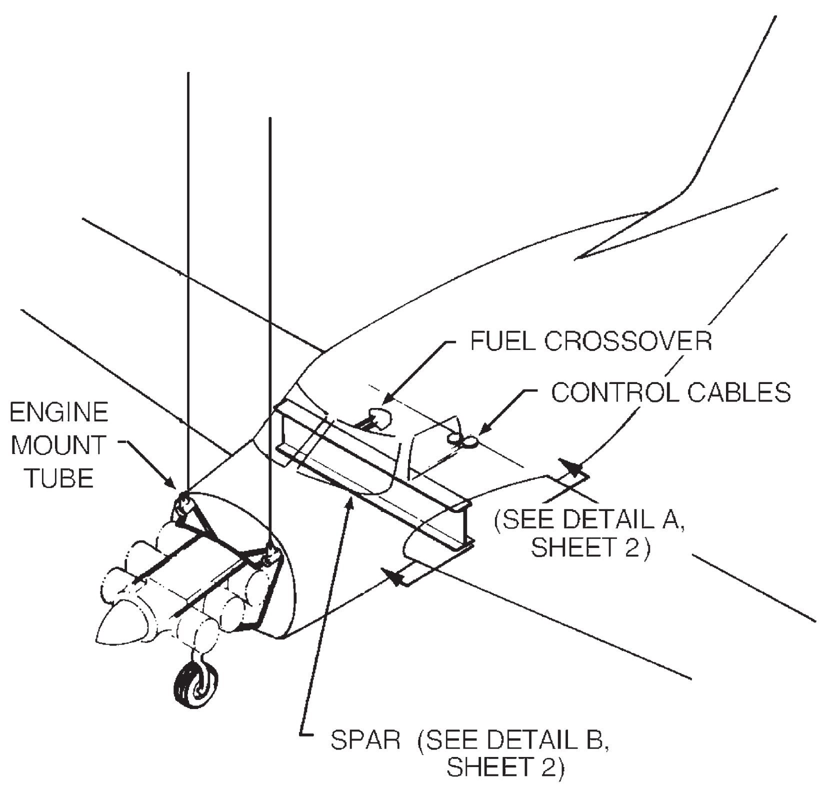
(a) Cradle to support fuselage: Create slings from industrial nylon lifting strap, six (6) inches wide. Lifting capability needed only if raising fuselage away from wing.

(b) Dolly to cradle wing: A single jack may be specially adapted for this procedure, in order to lower wing away from fuselage.

(c) Hoist assembly: To be attached to ceiling, with sufficient weight-bearing capacity. Position clamps so that straps will cradle fuselage at bulkheads (see “Figure 4”) for best support.

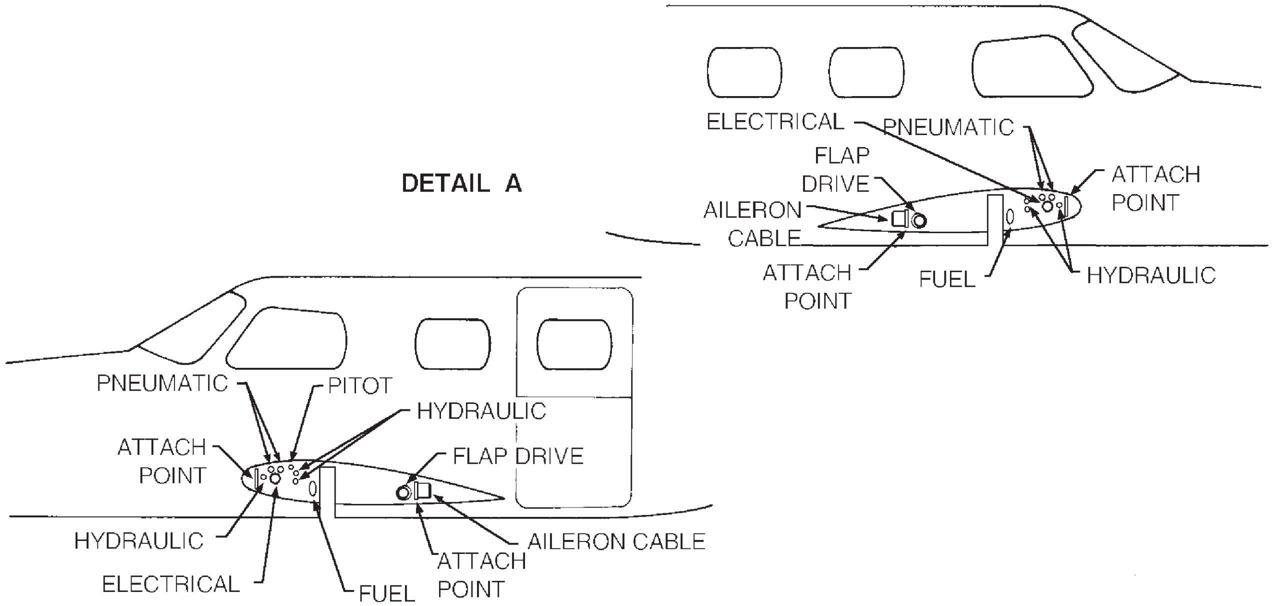
(d) Lifting pins (left and right) and cable: Suitable to attach to engine mount for additional steadying.

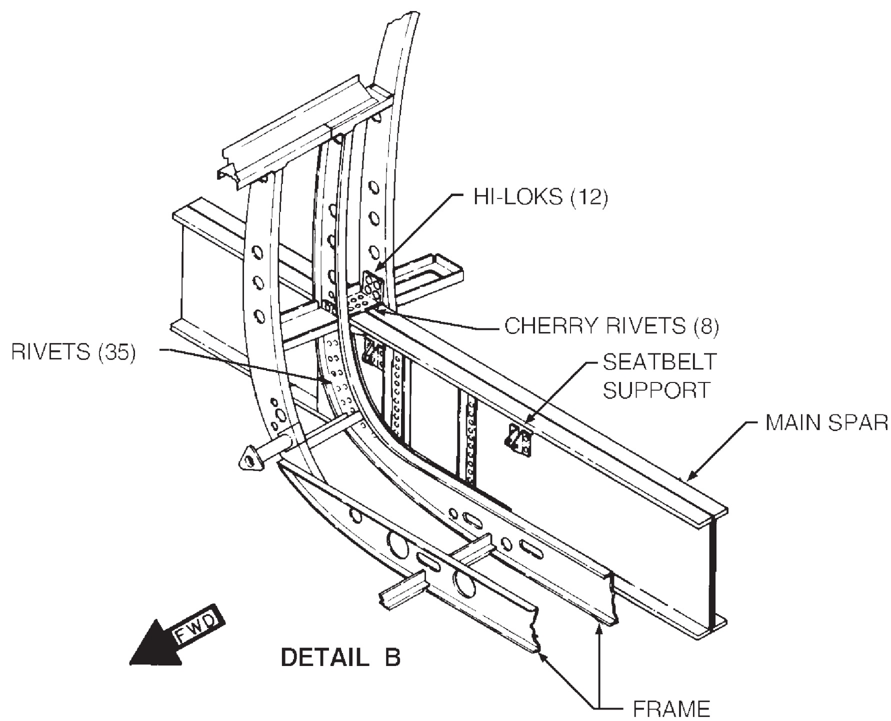
(e) Tools for removing rivets and cutting sealant: May be standard or fabricated.



Removal and Installation of Inboard Wing
Figure 5 (Sheet 1 of 2)

DETAIL A





DETAIL B

Removal and Installation of Inboard Wing
Figure 5 (Sheet 2 of 2)

B. Removal

(1) Remove all fuel from the wing.

(2) Prepare hoist and fuselage supports.

(3) Set up dolly under wing.

NOTE: Perform steps (4) thru (6) inside airplane.

(4) Remove interior furnishings as follows:

(a) Forward and center seats, left and right.

(b) Cabinets, left and right.

(c) Floor carpet.

(d) Forward side panels below window line, left and right.

(5) Remove floor panels from forward cockpit area to entrance door (F.S. 117.00 to F.S. 89.60).

(6) Remove lines and cables as follows:

(a) Remove relief tube, located under center left floorboard.

(b) Derig aileron, rudder, and stabilator cables. Remove from below spar.

NOTE: Do not disconnect hydraulic lines at this time.

(c) Remove fuel crossover valve cover and close-out, located on right side. Remove fuel crossover lines across front of spar. Remove fuel line (tear drop) crossover cover on left side.

(d) Remove fuel crossover valve.

NOTE: Perform steps (7) and (8) outside airplane.

(7) Remove wing-to-fuselage fairings, left and right wing roots. Remove bottom fuselage skin panels at wing spar, left, center, and right. Remove tee channels and angles below spar.

(8) Disconnect/remove lines as follows (“Figure 5, Detail A):

(a) Disconnect flap drive. Remove rods, left and right (see Figure 2, 20-00-00).

(b) Pull aileron cables through wing root and secure cables (refer to 27-00-00).

(c) Clear fuel lines from wing root area.

(d) Disconnect electrical cannon plugs, left and right.

NOTE: Perform step (9) with one man inside airplane at wing spar and one outside assisting. Use special tools to remove rivets and Hi-Loks.

(9) Remove rivets and hardware (see “Figure 5”, Detail B).

(a) Remove pilot seat belt supports, each side of forward spar

(b) Remove six forward and six aft Hi-Loks, each side, above spar.

(c) Remove five rivets thru fuselage skin and angle, each side, above spar.

(d) Remove sealant between fuselage, frame, and angle, each side, above spar.

(e) Remove floor board clips attached to aft spar.

NOTE: It may be necessary to cut sealant several times to get complete separation. Sealant after 28 days has 260 psi adhesion.

(10) Break sealant loose, forward and aft of spar each side.
  (a) Use a sharp putty knife and cut away excess sealant top and bottom of spar.
  (b) Use piano wire between spar and frame, pulling from inside and outside thru seals, like a saw. The wire will cut the sealant but not the aluminum. Drip PSS solvent on piano wire as you cut and this will help separation.

NOTE: Perform the following steps to support the fuselage for wing removal. Use fabricated hoist, cradle, and lifting pin.

(11) Attach straps, as shown in “Figure 4”, at indicated fuselage stations.

CAUTION: PROTECT AIRPLANE FINISH BY PLACING CLOTH OR PADDING UNDER STRAPS.

(12) Anchor fabricated lifting pins to engine mount tubes to secure nose and prevent fuselage from rotating (see Figure 4).

WARNING: DO NOT ENTER AIRPLANE BEFORE ATTACHING LIFTING PINS TO KEEP FUSELAGE FROM ROTATING IN SLINGS.

(13) Suspend fuselage with hoist. Attach cable to lifting pins.

NOTE: Dolly should already be in place under wing.

NOTE: Perform step (14) both inside and outside airplane.

(14) Retract landing gear and remove hydraulic lines as follows:
  (a) Using hydraulic pressure, release gear down locks.
  (b) Retract and tie back main landing gear.
  (c) Drain off hydraulic pressure.
  (d) Remove hydraulic lines from wing root and below spar (“Figure 5”).

NOTE: The following final steps may require additional manpower.

(15) Remove forward and aft wing attach bolts, each side. If sealant is completely free, wing will be ready to remove.

WARNING: MAKE CERTAIN THAT FUSELAGE LIFTING DEVICE AND LIFTING STRAPS ARE STRONG AND POSITIONED PROPERLY. SPAR FITS TIGHTLY IN CANTED SPAR BOX IN FUSELAGE. EVEN AFTER SEALANT HAS BEEN CUT, A HEAVY AMOUNT OF LIFTING PRESSURE (TO BREAK SEALANT GRIP) MAY BE REQUIRED. TAKE EXTREME CARE NOT TO DAMAGE FUSELAGE AND WING.

(16) Additional pressure will be necessary to separate wing from fuselage. Proceed as follows:
  (a) Have one man on each wing tip to steady the airplane and apply downward pressure.
  (b) Observe spar area inside airplane.
  (c) Strike spar with dead blow mallet until wing is completely released.

(17) With wing resting on dolly, lift fuselage to clear spar or lower wing.

(18) Remove wing.

C. Installation

WARNING: FUSELAGE MUST BE CRADLED.

NOTE: When installing a “salvaged” wing or wing section from another aircraft, verify the Stall Warning Heat Control conforms to the required configuration per Service Letter No. 1261.

NOTE: When installing a “replacement” wing or wing section, perform Stall Warning Flight Test Procedure, 27-30-00, upon completion of wing installation.

(1) Clean surfaces and all parts to be installed using PSS or other suitable solvent. Pay special attention to frame area inside airplane where sealant was broken or rivets removed. Remove all old sealant residue. Remove all corrosion inhibiting surface paint. This will ensure good sealant grip.

(2) Place new wing on dolly and position under fuselage.

(3) Raise wing or lower fuselage over wing.

(4) Secure forward and aft wing attach bolts, each side. (See “Figure 6”)

(5) Align spar with frames. Drill holes. Align angles with top of spar and with frames each side. Mark holes thru angles in top of spar each side.

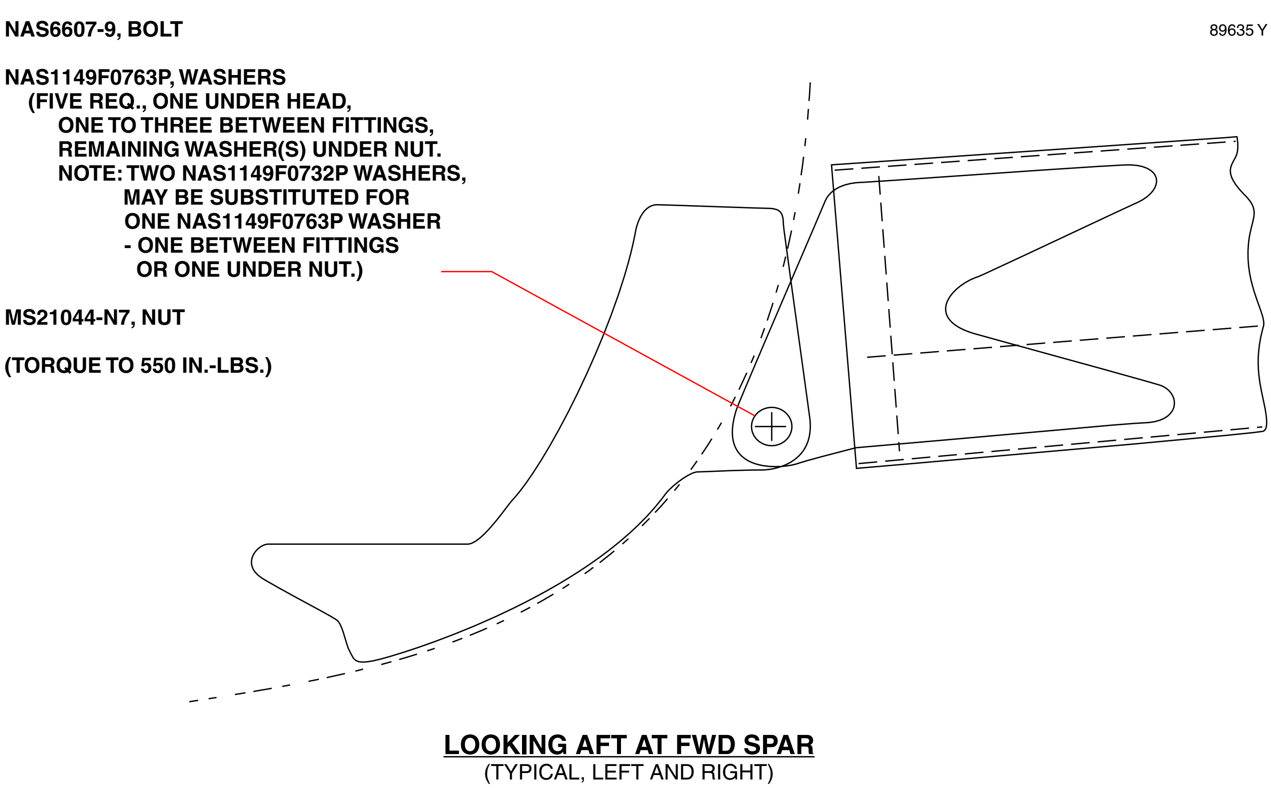
CAUTION: KEEP SURFACES FREE OF METAL SPLINTERS.

(6) Remove wing attach bolts and separate fuselage from wing. Move wing away from fuselage to work on each independently.

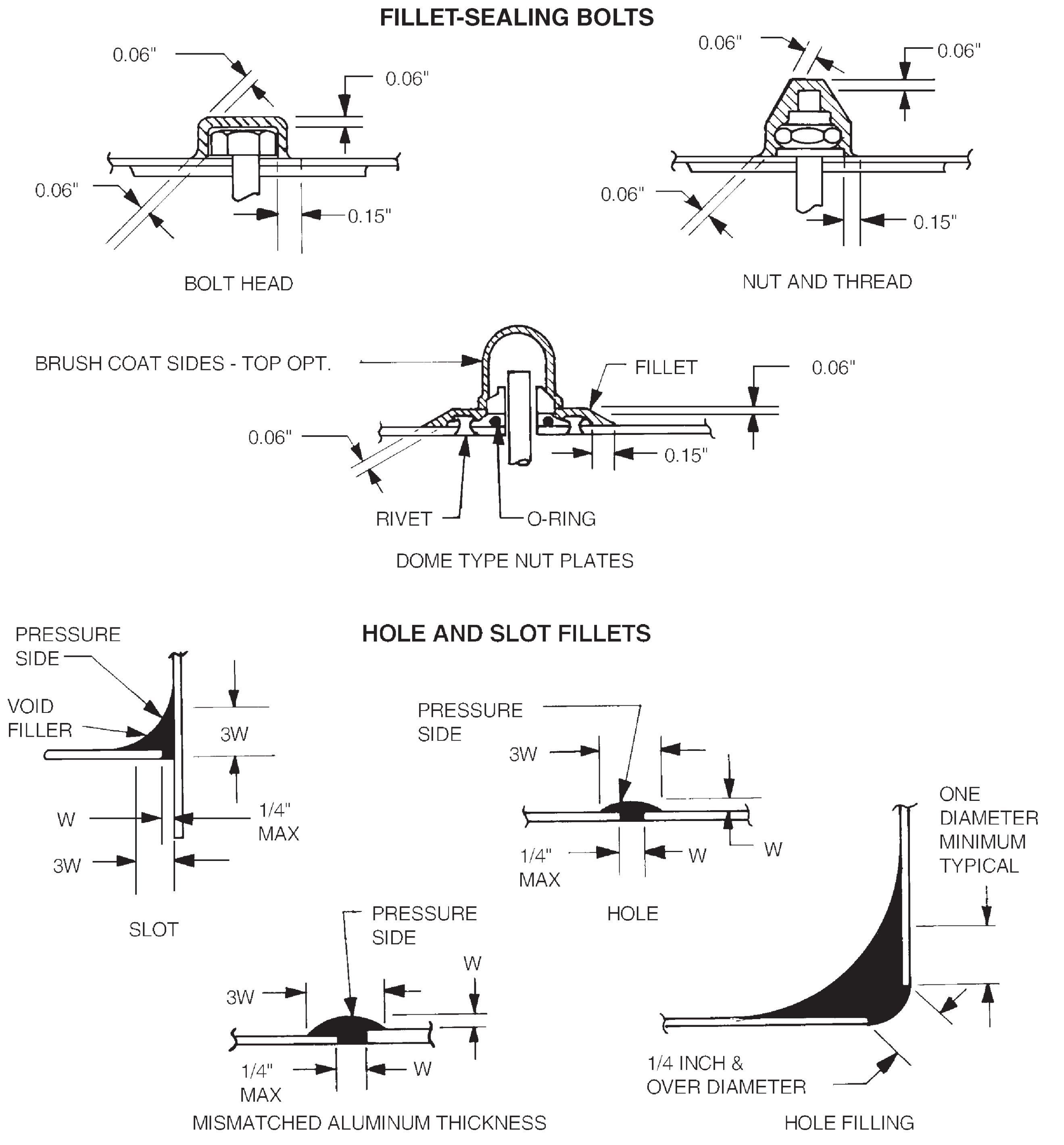
(7) Drill wing spar top each side for angles that attach to fuselage and frame.

(8) Use a vacuum or air hose to clean away all metal splinters in both wing spar and fuselage.

(9) Apply sealant to top spar and angles each side. Use R4 sealant.

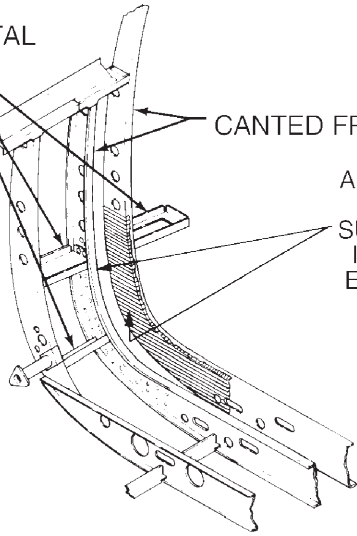


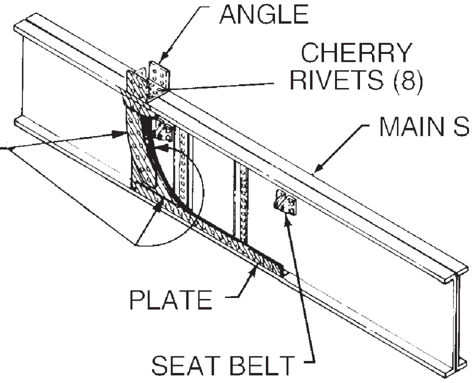
Installation of Inboard Wing
Figure 6



NOTE: Minimum fillet dimensions are shown. Fillet size shall be kept as near minimum as possible.

Hole and Slot Fillets
Figure 7





Application of Sealant
Figure 8

(10) Attach angles to top spar each side with cherry rivets. (Two #6/11, two #6/9, two #6/10 and two #6/11).

CAUTION: PROPER SEALING IS CRITICAL TO MAINTAIN PRESSURIZATION OF COCKPIT. BE GENEROUS WITH SEALANT PARTICULARLY IN CORNERS WITH WIDE GAPS. USE B4 SEALANT.

(11) Apply sealant to wing spar aft, forward, left and right plates and angles as shown in “Figure 7” and “Figure 8”.

(12) Apply sealant to canted frames as shown in “Figure 8”.

(13) Position wing under fuselage for installation.

(14) Lower fuselage onto wing spar. Some sealant will squeeze out.

(15) Secure wing with wing attach bolts. Install forward bolts, then aft Hi-Loks with nuts forward.

(16) Tighten forward nuts to a torque of 550 inch - pounds and tighten aft nuts (Hi-Loks) until they shear off.

NOTE: Two people are needed to buck and shoot rivets.

(17) Clean off excess sealant with PSS solvent.

(18) Rivet canted frames to left and right aft spar. (Ten #6/9, ten #5/8, and ten #5/7.)

(19) Rivet floorboard clips to aft spar.

(20) Rivet canted frame at left forward spar. (Ten #6/9, ten #5/8, six #5/7 and four #5/8 thru floorboard clip.)

(21) Rivet canted frame to spar at right forward spar. (Ten #6/9, seven #5/8, three #5/9 thru lower intercostal, six #5/7, and four #5/8 thru floorboard clip.)

(22) Install five (5) rivets in angle on top of spar each side thru fuselage skin from outside fuselage.

(23) Install four (4) Hi-Lok bolts thru each intercostal, frame, angle and two (2) thru frame and wing spar, just under intercostal. All four locations are left, right, fwd and aft of top wing spar. See “Figure 5” and “Figure 8”.

(24) Install four (4) angles that form two (2) tees in bottom of fuselage. Tap with a plastic mallet thru aft channels in the frame under wing spar until they clear the forward frame under spar. Then, using small Vise Grips pliers, attach to angles and tap through forward frame channels an equal amount.

(25) Inside fuselage align angles evenly with Kleco fasteners to existing holes one on each side of stringer.

(26) Below fuselage place short tee angles between floor angles and rivet together with four (4) rivets on top. Then go inside and rivet both ends to bottom stringer.

(27) Ensure there are no metal splinters or chips between skins prior to installing close-outs.

(28) Apply sealant to all skins and close-out skins on bottom of aircraft. See “Figure 6”. Apply sealant 1/4 inch each side of rivets.

(29) Install close-outs in place with Kleco fasteners and insert rivets from bottom.

(30) Inspect rivets.

(31) Install forward pilot’s seat belt brackets with bolts.

(32) Install cat grommets on all passage holes in bottom spar.

(33) Inside fuselage at spar all newly installed rivets and Hi-Loks must be brush sealed with A-4 sealant.

(34) Make adjustments as follows:

   (a) Control cables (refer to Aileron Control - Rigging and Adjustment, 27-10-00).

   (b) Fuel crossover lines and valve (refer to 28-20-00).

   (c) Hydraulic lines (refer to Hydraulic System Testing and Bleeding Hydraulic System, 29-10-00).

      1) Put pressure on system.

      2) Allow gear to extend.

      3) Test and bleed system.

      4) Perform gear cycle test (refer to Landing Gear Retraction System Functional Test, 32-30-00).

      5) Be sure that all three landing gear are down and locked.

(35) Remove support straps.

(36) Remove dolly.

(37) Install wing-to-fuselage fairings and bottom skin panels.

(38) Install interior furnishings.

(39) Reinstall lines and cables as follows: (“Figure 5”, Detail A)

   NOTE: Reinstall electrical, fuel and hydraulic lines first. Then, reinstall control cables.

   (a) Outside airplane.

      1) Install hydraulic lines in wing root and below spar.

      2) Connect electrical cannon plugs, left and right.

      3) Install fuel lines in wing root area.

      4) Pull aileron cables through wing root and secure cables (refer to 27-00-00).

      5) Connect flap drive. Reinstall rods, left and right (see Figure 2, 20-00-00).

(b) Inside airplane.

1) Reinstall fuel crossover valve.

2) Reinstall fuel crossover lines across front of spar.

3) Reinstall fuel line (tear drop) crossover cover on left side.

4) Reinstall fuel crossover valve cover and close-out, located on right side.

5) Reconnect aileron, rudder, and stabilator cables. Rig per Chapter 27.

6) Reinstall relief tube, located under center left floorboard.

5. Attach Fittings Corrosion Control

Each five years, inspect the attach fittings shown in “Figure 9” for corrosion and condition. Repair/replace as required. When inspection and repair/replacement is completed, reapply Dinitrol AV 8 (P/N 89500-800) Corrosion Inhibiting Compound.

NOTE: For all fittings, inspection of the exposed portion is sufficient unless obvious or extensive corrosion is apparent.

NOTE: Any part exhibiting flaking of the metal due to rust must be replaced.

NOTE: If aluminum skin or structure is corroded (beyond a light surface oxidation), the area must be repaired or replaced.

NOTE: On any part being replaced, dip rivets in fluid resistant epoxy primer or equivalent and install wet.

A. When inspection and repair or replacement are completed, allow sufficient time for primer to dry, and apply Dinitrol AV 8 to:

(1) Forward wing spar attach fittings.

(2) Aft wing spar attach fittings.

NOTE: Includes wing and fuselage fittings.

CAUTION: READ PRECAUTIONS ON CONTAINER LABEL AND WEAR A CHARCOAL FILTER MASK WHEN APPLYING DINITROL AV 8.

B. Apply Dinitrol by spraying or brush. Use an air or airless spray gun if available. A hand aspirator bottle may be used if a sufficient fan spray pattern can be developed. Product should not be thinned.

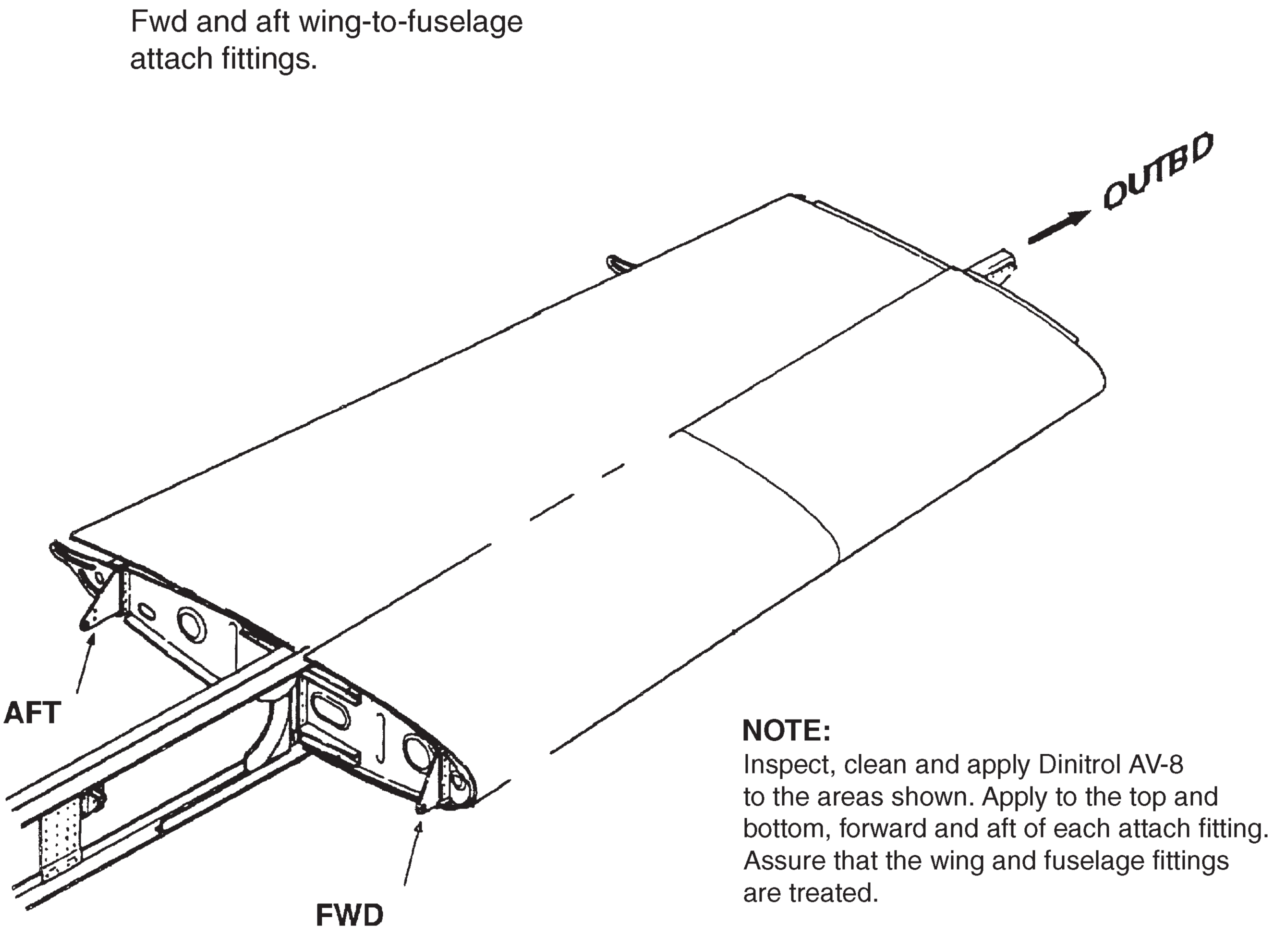
C. Cover autopilot servos, electrical connectors, and cable pulleys.

WARNING: DO NOT GET DINITROL INTO AUTOPILOT SERVOS OR CLUTCHES.

D. Spray directly into faying surfaces and mating points.

E. Ensure entire attach fitting is treated. Spray inside where the fitting is attached to the structure.

F. Apply to approximately .8 mils in thickness. Avoid puddling and running. Allow to dry 1 to 1 1/2 hours before airplane use. Full drying takes 6 to 8 hours.



Wing Attach Fittings Corrosion Control
Figure 9

6. Wing / Flap Protective Tape

S/N's 4636001–4636768 and 4692001–4692214 had protective tape installed either on the underside of the wings or the upper surface of each flap to prevent chaffing between the flaps and the underside of the wings as the flaps were extended and retracted. See Piper Service Letter 1233, per latest revision, to remove tape from the wings or flaps and install an improved tape, with an improved service life, on the underside of each wing.

The following instructions address the removal and replacement of tape on the underside of the wings in S/N's 4636769 and up, and in earlier serial numbers after SL1233 has been accomplished.

A. Preparation

(1) Lower the flaps to gain access to the flap and wing skin surfaces.

(2) Set the battery master switch to the OFF position.

B. Removal

(1) Locate the protective tape on the underside of the upper wing skin.

(2) Remove the existing protective tape (or tape remnants) from the wing, slowly peeling it away to avoid any damage to the wing surface.

(3) Prepare a cleaning cloth with isopropyl alcohol, wearing protective gloves.

CAUTION: WIPE, DO NOT RUB, AS IT COULD RESULT IN RUBBING DIRT INTO THE WING SURFACE.

(4) Clean the surface of the wing where the protective tape was installed. Wipe the cleaning cloth in one direction only.

(5) Using a second clean cloth, remove any remaining isopropyl alcohol from the wing surface before it dries.

(6) Repeat Steps (3) through (5) until the wing skin is clean from all dirt, soil, and adhesive residues. Use additional cloths and isopropyl alcohol as necessary.

C. Installation

As shown in Figure 10 on page 574020, the new Teflon tape (Piper P/N 380-066) will be installed on the lower surface (or, underside) of the upper wing skin, aligned with the edge of the wing skin.

(1) General

(a) Work on one wing at a time.

(b) This tape should only be installed when ambient temperatures are above 60 °F (15.6° C).

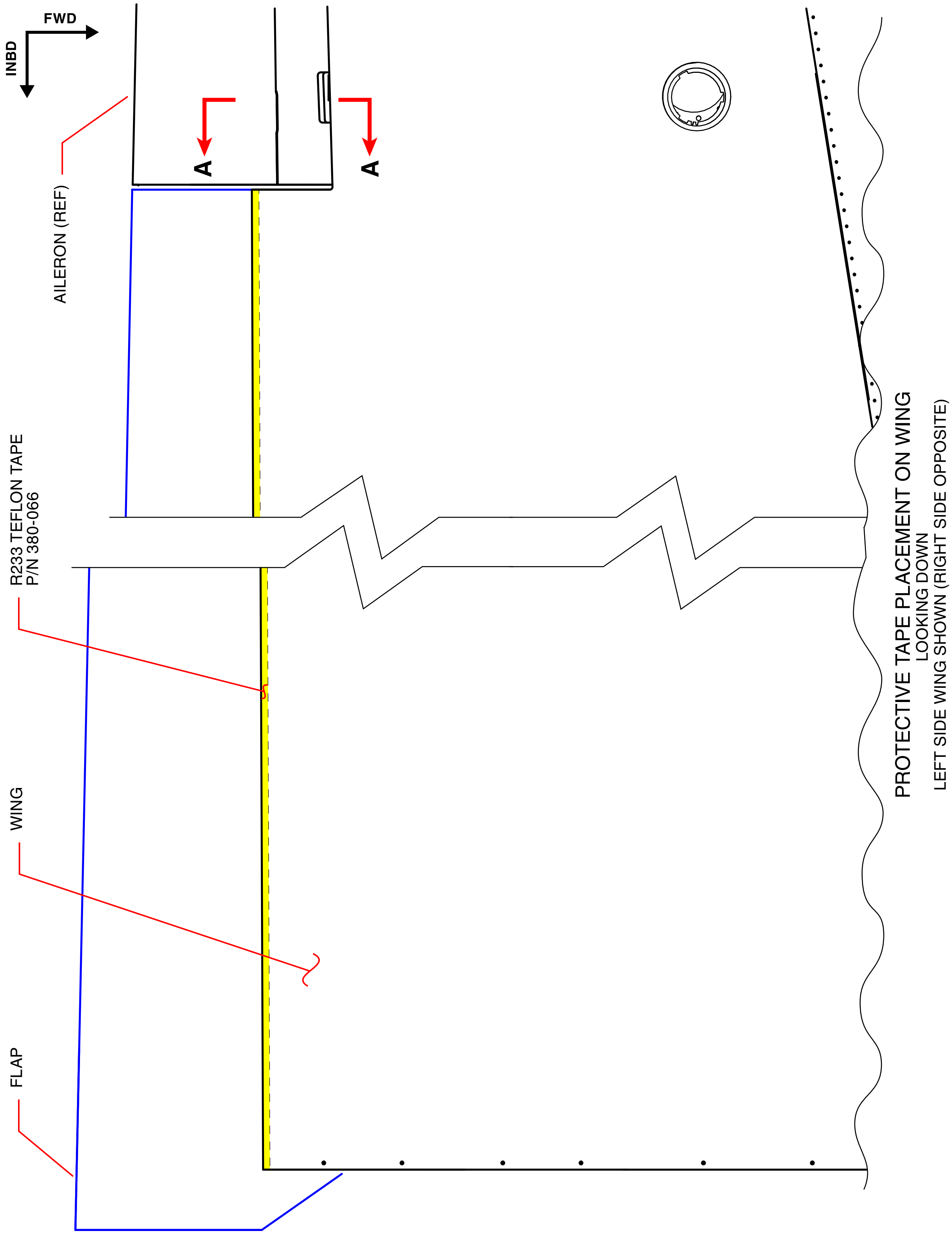
(c) If wing has been repainted, ensure paint has cured for a minimum of 24 hours prior to installing tape.

(d) Confirm that the tape has not exceeded its shelf life prior to installation.

(e) Do not clean surfaces in preparation for installing the Teflon tape while significant dust and dirt is present in the work area.

(f) If the Teflon tape is not applied within 30 minutes after cleaning the wing surface, the wing must be recleaned.

(g) Clean hands thoroughly before handling the tape roll. Hands and tape must be clean during the installation to ensure maximum adhesion. Do not use tape that shows signs of being soiled or contaminated.



Protective Tape Installation on Wing Skin
Figure 10

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(2) Pre-Installation Cleaning

(a) Prepare a cleaning cloth with isopropyl alcohol, wearing protective gloves.

CAUTION: WIPE, DO NOT RUB, AS IT COULD RESULT IN RUBBING DIRT INTO THE WING SURFACE.

(b) Clean the surface of the wing where the protective tape is to be installed. Wipe the cleaning cloth in one direction only.

(c) Using a second clean cloth, remove any remaining isopropyl alcohol from the wing surface before it dries.

(d) Repeat Steps (b) through (c) until the wing skin is clean from all dirt, soil, and isopropyl alcohol as necessary.

(3) Installation

(a) Peel off the backing from a 12 in. length of tape.

(b) Position the tape, aligned with the edge of the wing skin. Press firmly to the surface with fingers.

NOTE: Do not allow wrinkles or creases to form when applying tape.

(c) With a plastic applicator, roller or squeegee, apply the remainder of the length of tape using firm, overlapping strokes along the length and width of the tape.

(d) Once the length of tape has been applied, peel back another 12 in. length of tape.

(e) Repeat Steps (b) through (d) until tape has been installed on the wing skin along the entire length of the flap.

(f) If required, without cutting into the wing surface, trim the tape flush to the wing skin with a sharp razor blade.

(4) Finishing

Allow the tape to dwell on the wing skin for at least 24 hours before raising the flaps, to ensure proper adhesion. In 24 hours, tape will reach 80 percent adhesion; maximum adhesion is achieved in 72 hours.

D. Functional Test

(1) Have two assistants apply gentle upward pressure on the aft edge of each flap while cycling the flaps several times. While cycling each flap, make sure to observe the Teflon tape for any damage, such as tearing, ripping, or peeling.

(2) Without any upward pressure on the aft edge of the flaps, cycle the flaps several times. While cycling each flap, make sure to observe the Teflon tape for any damage such as tearing, ripping, or peeling.

FLIGHT SURFACES

1. Aileron

See “Figure 1”.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

A. Removal

(1) Remove the nut and bolt which secures the aileron drive sector to the yoke in the aileron.

(2) Support the aileron and remove the two aileron hinge bolts.

(3) Remove the aileron.

B. Installation

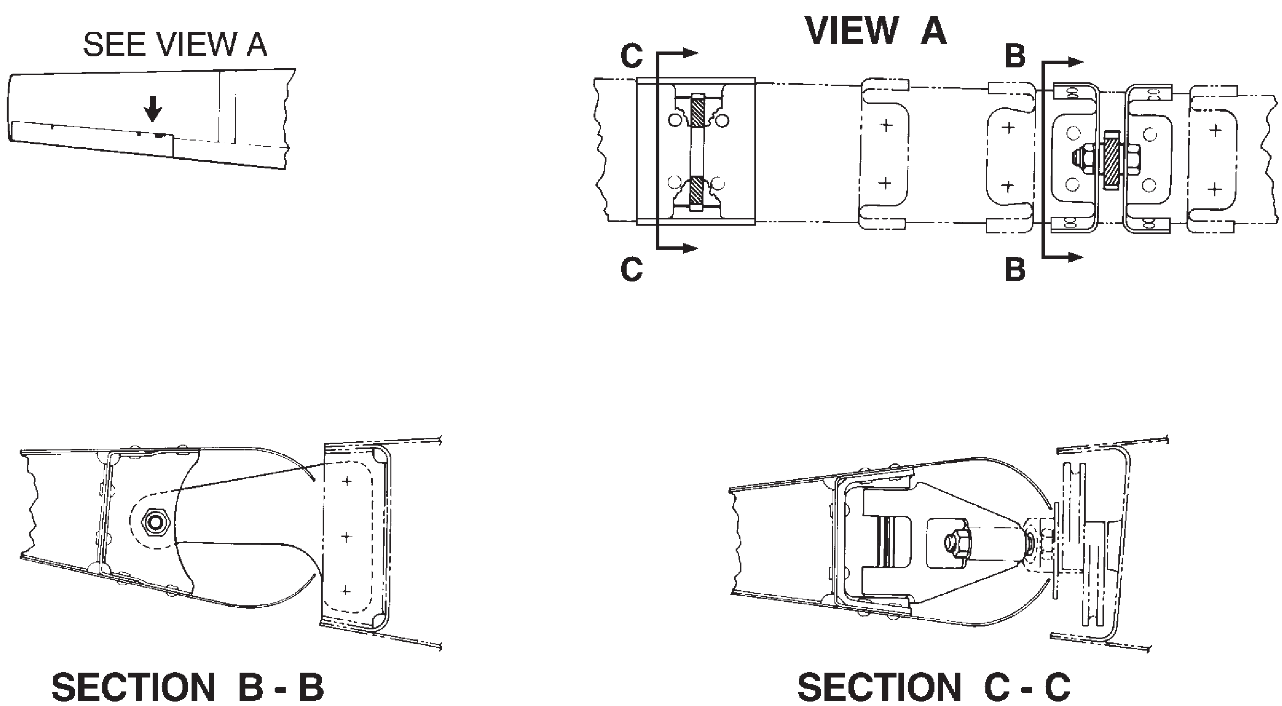
WARNING: AILERONS THAT HAVE BEEN REPLACED OR REPAINTED MUST BE BALANCED BEFORE INSTALLATION. SEE BALANCING, BELOW.

(1) Position the aileron on the trailing edge of the wing.

(2) Install the attaching nuts, bolts and washers.

(3) Attach the aileron drive sector to the yoke in the aileron with the previously removed nut, bolt and washers.

(4) Actuate the aileron controls to ensure freedom of movement.



Aileron Installation
Figure 1

C. Balancing

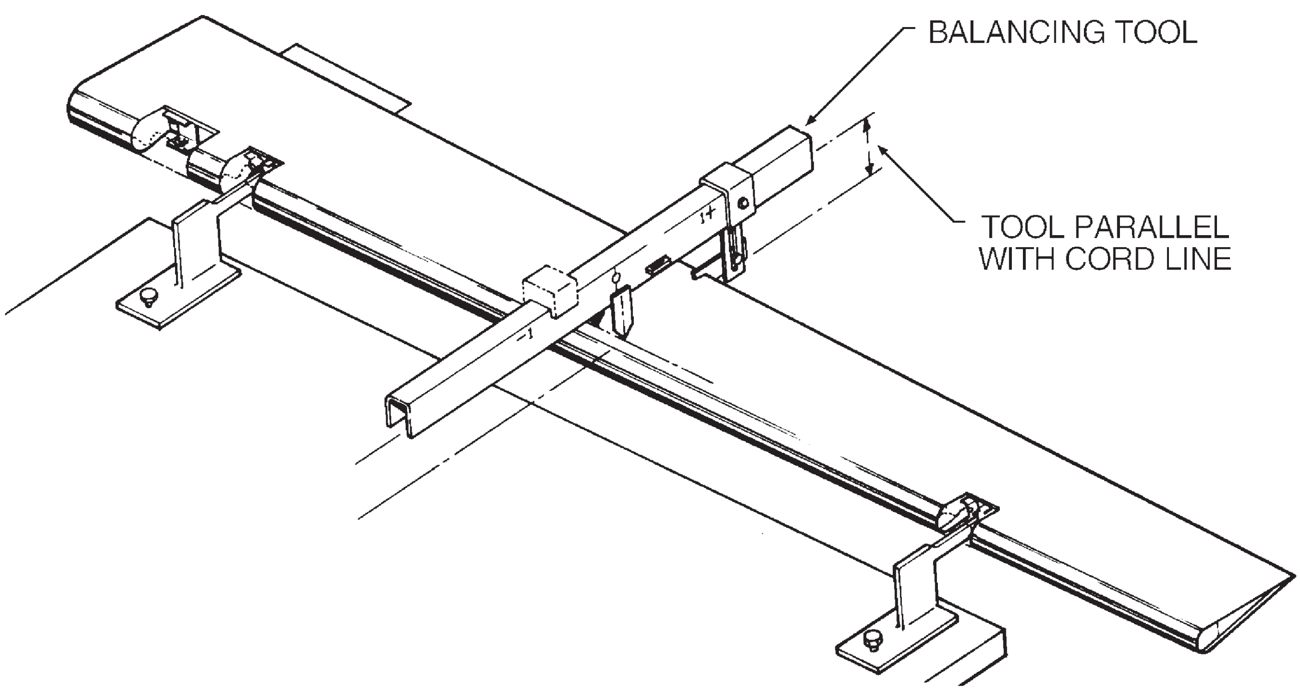
See “Figure 2”.

To balance the aileron, the assembly must be complete and in its final flight configuration as specified under Balancing Equipment in 55-20-00 (including all attaching screws). Before balancing, bend the trim tab into its neutral position. Place the aileron assembly on the knife edge supports in a draft free room with the beam perpendicular to the hinge centerline. Do not place the tool on the trim tab. Calibrate the tool as described in Balancing Equipment, 55-20-00. Read the scale when the bubble level has been centered by adjustment of the movable weight and determine the static balance limit (see “Chart 1”).

There is no adjustment provision for the ailerons.

CHART 1
AILERON STATIC BALANCE LIMITS

Leading Edge Heavy (Inch-Pounds)	Trailing Edge Heavy (Inch-Pounds)
0	-5.2



Balancing Aileron
Figure 2

2. Wing Flaps

See “Figure 3”.

CAUTION: CONTROL SURFACE SKINS MUST BE REPLACED IF THEY SUSTAIN DAMAGE OR EXHIBIT CRACKS.

A. Removal

(1) Extend the flap and disconnect the flap drive push rods (2 places each flap).

(2) Support the flap and remove the flap hinge bolts, nuts and washers.

(3) Pull the flap straight back off the wing.

B. Installation

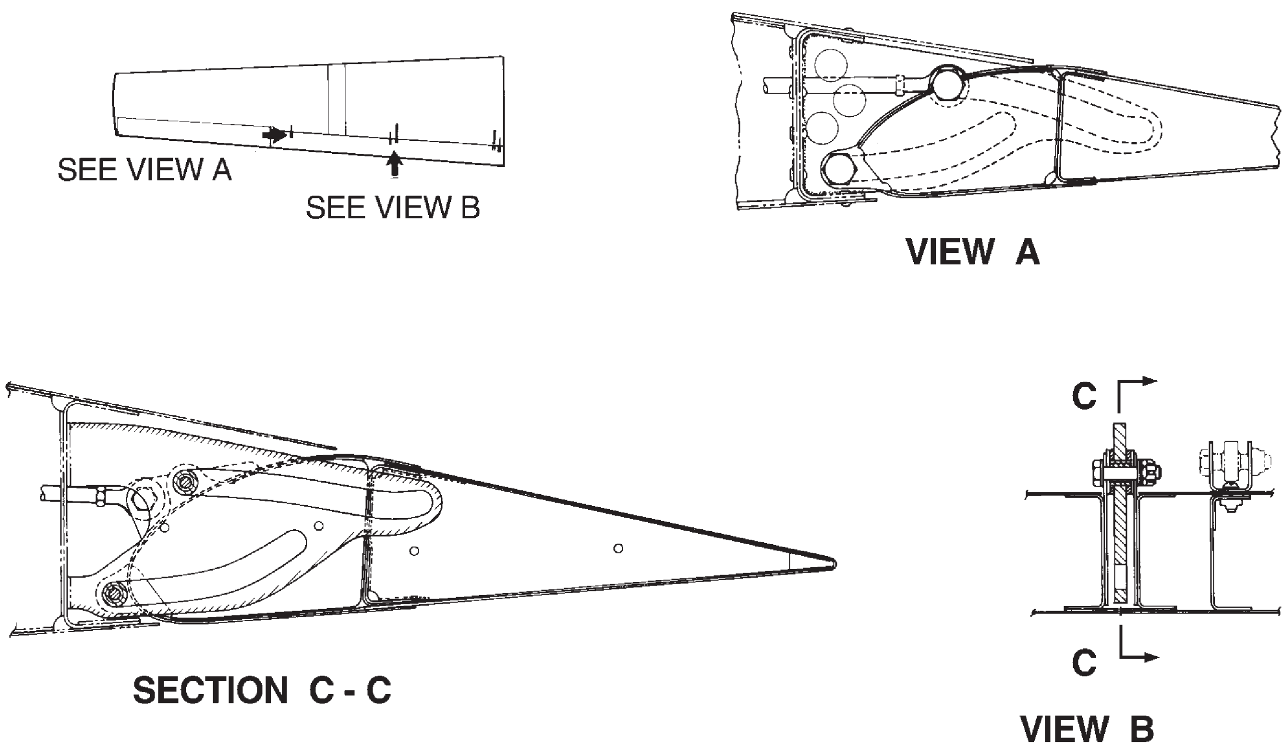
(1) Position the flap on the trailing edge of the wing.

(2) Ensure that spacers are in place in the flap tracks.

(3) Insert bolts, washers and nuts. Torque the nuts from 38–43 inch pounds. Install the teflon faced washers on the center tracks so that the teflon side faces the flap track.

(4) Connect the two flap drive push rods to the aileron.

(5) Cycle the flaps several times to make sure that they are operating freely (refer to Flap Rigging and Adjustment, 27-50-00).



Wing Flap Installation
Figure 3

CHAPTER

61

PROPELLER

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CHAPTER 61

LIST OF EFFECTIVE PAGES

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CHAPTER 61 - PROPELLER

TABLE OF CONTENTS

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GENERAL

Description

(PIR-PS50077, Rev. CV.)

The PA-46-350P Malibu Mirage/M350 may be equipped with a Hartzell, two-bladed, all-metal, constant-speed propeller, a Hartzell, three-bladed, Kevlar-composite, constant-speed propeller, or a Hartzell, three-bladed, Carbon-composite constant speed propeller. Each blade model has an 80 inch diameter.

The PA-46R-350T Malibu Matrix may be equipped with a Hartzell, three-bladed, Kevlar-composite constant-speed propeller or a Hartzell three-bladed, Carbon-composite constant speed propeller. Each blade model has an 80 inch diameter.

In all installations, the propeller is controlled by an engine-mounted Hartzell governor which supplies oil through the propeller shaft. Oil pressure increases the blade pitch (reduces rpm) and blade centrifugal force reduces the blade pitch (increases rpm).

When propeller heat is installed, a single-element system is used, as described in section 30-60-00.

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PROPELLER ASSEMBLY

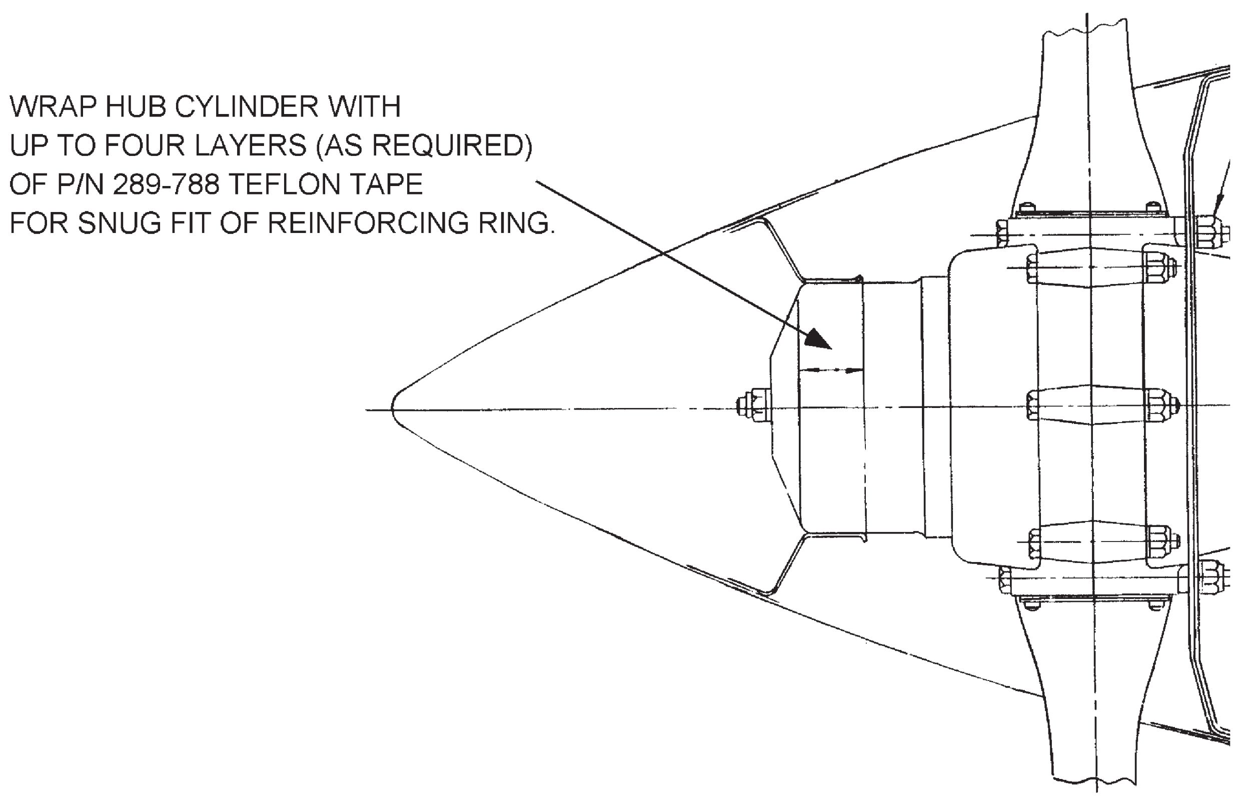
1. Propeller

A. Removal

(1) Ensure master and magneto switches are off.
(2) Move fuel selector to off position.
(3) Place mixture control in idle cut-off.
(4) Note position of each component to facilitate reinstallation.
(5) Remove the screws from around the spinner assembly and remove the spinner.
(6) Place a drip pan under the propeller to catch oil spillage.
(7) Remove the six propeller mounting bolts and remove propeller.

B. Installation

(1) Ensure master and magneto switches are off.
(2) Move fuel selector to off position.
(3) Place mixture control in idle cut-off.
(4) Observe the starter ring gear to make sure it is mounted properly on the engine crankshaft flange. One of the bushings on the crankshaft is stamped with an “O” mark and must be inserted in the starter ring gear hole, likewise identified with an “O” mark.



Spinner Installation
Figure 1

(5) Wipe crankshaft and propeller pilot to ensure that no chips or foreign matter enter the propeller mechanism.

(6) Check interior of propeller hub for proper seating of O-ring. Wipe inside of hub to remove any traces of dirt. Check to see that O-ring is covered with grease.

(7) Install propeller with blades aligned with mounting bolt hole marked “O”.

(8) If not installed already, install rear spinner bulkhead using four (4) or six (6) hub clamping bolts, two and three blade installations, respectively. Torque 20–22 ft.-lbs., or as specified in the appropriate Hartzell Propeller Owner’s Manual (i.e., 115N or 145) latest revision.

(9) Slide propeller carefully over pilot, taking care that O-ring is not damaged.

(10) Install the six propeller mounting bolts and torque 60–70 ft.-lbs., or as specified in the appropriate Hartzell Propeller Owner’s Manual (i.e., 115N or 145) latest revision.

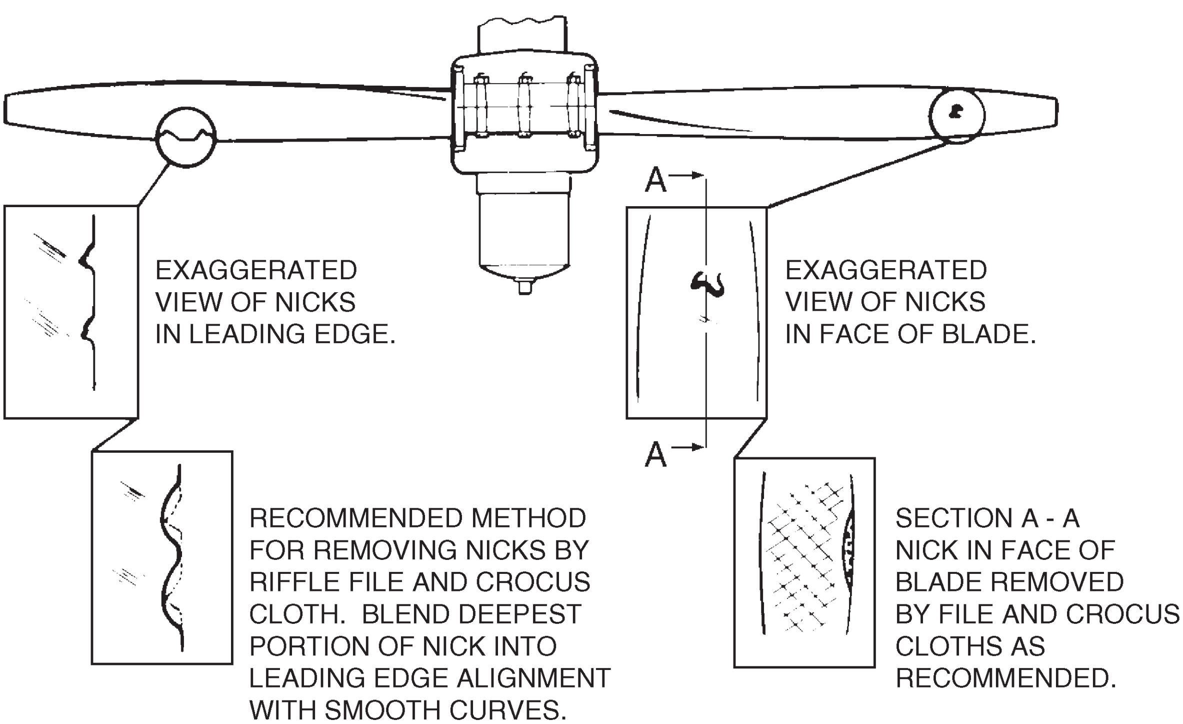
(11) Check propeller blade track.

(12) Safety the propeller mounting bolts with safety wire.

(13) Grease propeller per Lubrication Chart in 12-20-00.

(14) Wrap hub cylinder as required with 289-788 Teflon tape. Refer to “Figure 1”.

(15) Install spinner.



Typical Nicks and Removal Method for All-Metal Blades
Figure 2

C. Blade Track

Blade track is the ability of one blade tip to follow the other, while rotating, in almost the same plane. Excessive difference in blade track - more than 0.0625 inch - may be an indication of bent blades or improper propeller installation. Check the blade track as follows:

(1) With the engine shut down and blades vertical, secure a smooth board to the airplane just under the tip of the lower blade. Move the tip fore and aft through its full blade-shake travel, making small marks with a pencil at each position. Then center the tip between these marks and scribe a line on the board for the full width of the tip.

(2) Carefully rotate the propeller by hand to bring the opposite blade down. Center the tip and scribe a pencil line as before and check that the lines are not separated more than 0.0625 inch.

(3) Propellers having excess blade track should be removed and inspected for bent blades, or for parts of sheared O-rings, or foreign particles, which have lodged between the hub and crankshaft mounting faces. Bent blades will require repair and overhaul of assembly.

D. Cleaning, Inspection, and Repair

(1) Check for oil and grease leaks.

(2) Clean the spinner, propeller hub interior, exterior, and blades with a non-corrosive solvent.

(3) Inspect the hub parts for cracks.

(4) Steel hub parts should not be permitted to rust. Use aluminum paint to touch up if necessary, or replate during overhaul.

(5) Check all visible parts for wear and safety.

(6) Check the blades to determine whether they turn freely on the hub pivot tube. This can be done by rocking the blades back and forth through the slight freedom allowed by the pitch change mechanism. If they appear tight and are properly lubricated, remove the pitch change mechanism so that each blade can be checked individually. If the blades are tight, the propeller should be disassembled.

2. Propeller Blade Inspection and Repair

A. All-Metal Blades

(1) Inspect the blades for damage or cracks. Nicks in the leading edges of the blades should be filed out and all edges rounded, to prevent formation of cracks. Use fine emery cloth for finishing.

(2) Refer to “Figure 2” for all-metal propeller blade care.

B. Composite Blades

CAUTION: SEE LATEST REVISION HARTZELL MANUAL NO. 135, MAINTENANCE MANUAL FOR COMPOSITE PROPELLER BLADES, FOR SPECIFIC DAMAGE LIMITS AND APPROVED REPAIR PROCEDURES. THE FOLLOWING INFORMATION IS ADVISORY ONLY.

(1) Description.

The composite blade is composed of a metal blade shank retention section into which is molded a low-density foam core that supports built-up layers of composite laminate. An erosion shield of electroformed nickel is adhesively bonded to the blade leading edge to protect it from impact damage. A finish covering of polyurethane paint protects the entire blade from erosion and ultraviolet damage.

(2) Inspection.

Inspect entire blade for nicks, gouges, looseness of material, erosion, cracks and debonds, and lightning strikes. Each 500 hours coin tap the erosion shield. Each 1000 hours coin tap the entire blade surface. Defects or damage discovered during inspection must be evaluated in accordance with the following criteria to determine: if repairs are required prior to further flight; and, if so, whether or not repairs may be conducted locally or must be performed by a qualified propeller shop.

Coin-Tap Test:

(a) Composite blades can be inspected for delaminations and debonds by tapping the blade, or cuff (if applicable), with a “metal washer.”

(b) Use a washer-shaped metal tapper, approximately 2.5 inches O.D. x 1.25 inches I.D. x 0.25 inch thick, and weighing no less than 3 oz. Tap the surface. If an audible change is apparent, sounding hollow or dead, a debond or delamination is likely.

NOTE: Blades which incorporate a “cuff” will have a different tone when coin-tapped in the cuff area. To avoid confusing sounds, the cuff area and the transition area between cuff and blade should be coin-tapped separately from the blade area.

(c) Mapping of the area being inspected is recommended. Coin-tap within an imaginary grid or matrix consisting of 2 inch squares during scheduled aircraft inspections. During blade overhaul, a more thorough inspection is required by using a smaller, 0.5 inch, grid.

(d) The metal erosion shield is more likely to have problems than the blade, therefore a more thorough coin-tapping of the erosion shield is desirable. Also, slight deformations in the erosion shield may be noticed by careful visual and manual (touch) inspection. Such deformations may be the result of a debond and should be given a careful coin-tap inspection.

(e) If a suspected delamination or debond is discovered, a localized, thorough coin-tap inspection is required to define the precise area of delamination or debond.

(f) Outline any suspect areas with a grease pencil to determine approximate size of damage. If repairable, proceed per Hartzell Manual No. 135, latest revision.

(3) Definitions.

Airworthy Damage	Damage that does not affect the safety or flight characteristics of the propeller blade. Although a blade may continue in service with airworthy damage, this type of damage should be repaired at the earliest practical time to prevent further damage to the blade.
Unairworthy Damage	Damage that exceeds the maximum limits of airworthy damage. Unairworthy damage can affect the safety or flight characteristics of the propeller blade. This type of damage must be repaired at prior to the next flight. Exceptions may be possible, but require written authorization from Hartzell.
Minor Repair	Correction of damage that may be safely performed in the field by a certified aircraft mechanic (preferably one who has completed Hartzell composite blade training).
Major Repair	Correction of damage that cannot be performed by elementary operations. Major repairs must be performed by a Hartzell Propeller Inc.-approved repair station. Exceptions may be possible, but require written authorization from Hartzell.

(4) Airworthy Damage Limits.

CAUTION: SEE LATEST REVISION HARTZELL MANUAL NO. 135, MAINTENANCE MANUAL FOR COMPOSITE PROPELLER BLADES, FOR SPECIFIC DAMAGE LIMITS.

(5) Minor Repairs.

CAUTION: SEE LATEST REVISION HARTZELL MANUAL NO. 135, MAINTENANCE MANUAL FOR COMPOSITE PROPELLER BLADES, FOR SPECIFIC DAMAGE LIMITS AND APPROVED REPAIR PROCEDURES. THE FOLLOWING INFORMATION IS ADVISORY ONLY.

Procedures for the following minor repairs are provided in latest revision Hartzell Manual No. 135 - Maintenance Manual for Composite Propeller Blades.

(a) Nickel Erosion Shield.

-- Debond extending to trailing edge or crack.

-- Gouge.

-- Missing area of trail side or inboard end.

(b) Blade Cuff.

Cracks at the root end of cuff.

-- Nick or scratch.

-- Cracks.

-- Delamination.

(c) Composite Blade.

-- Gouges or loss of material.

-- Crushed trailing edge.

-- Split trailing edge.

-- Frayed trailing edge.

(d) Lightning Strike.

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CONTROLLING

1. Propeller Governor

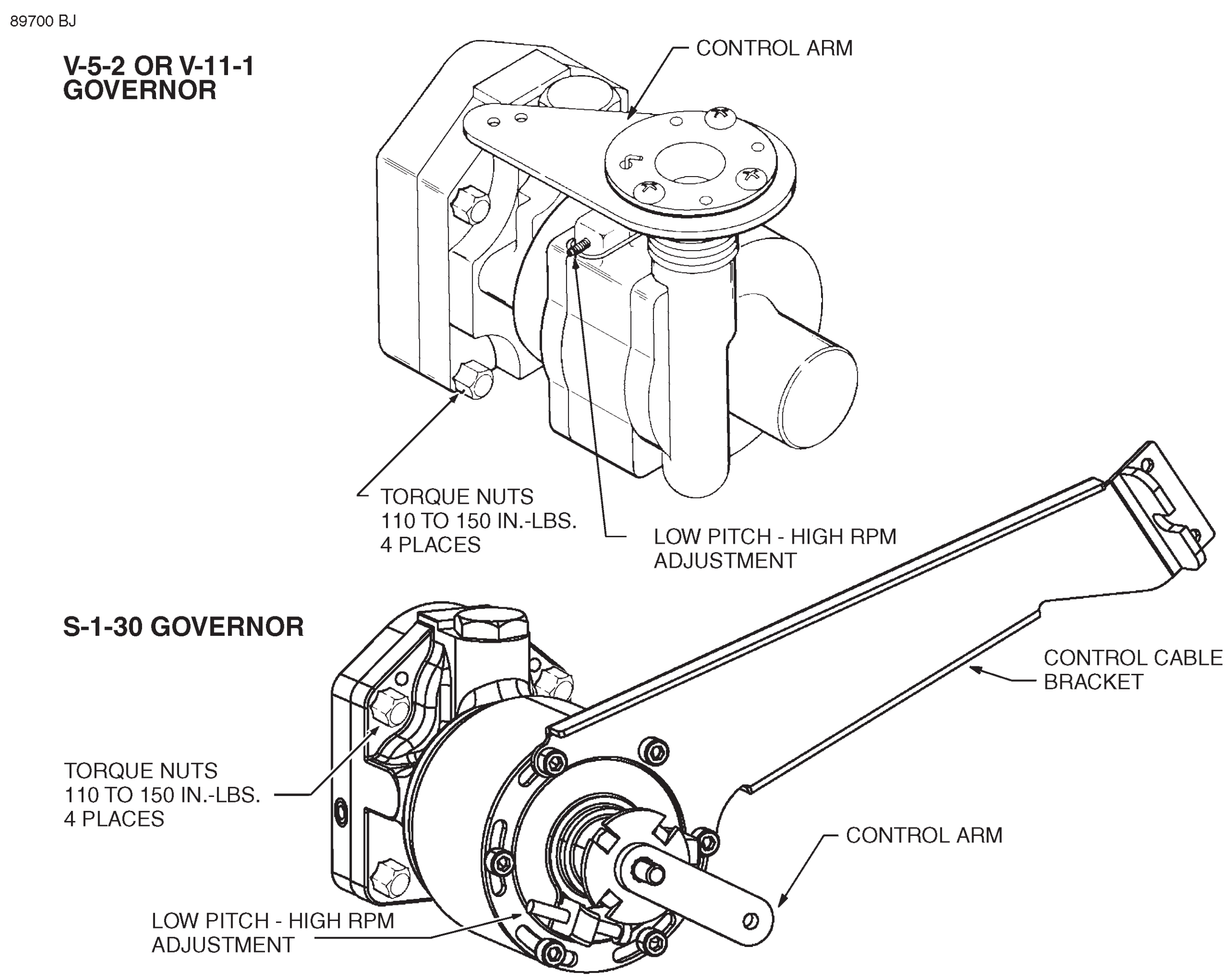
   A. Removal

   The propeller governor is mounted on the upper forward portion of the crankcase. Remove the governor as follows:

   (1) Remove the upper cowl and left side of the nose cowl to gain access to the governor.

   (2) Disconnect the governor control cable end from the governor control arm.

   (3) Remove the governor mounting nuts and withdraw the governor, with control bracket, from the mounting pad. Cover the mounting pad to prevent foreign material from entering the engine.



Propeller Governor
Figure 1

B. Installation

(1) Clean the mounting pad and the governor drive shaft thoroughly.

(2) Coat the mounting gasket with Dow Corning release agent or equivalent.

(3) Lubricate the drive shaft with engine oil and install the governor, with control mounting bracket. on the mounting pad.

(4) Tighten the mounting bolts evenly and tighten to a final torque as shown in “Figure 1”.

(5) Connect the control cable to the control arm. Check to be sure the attachment bolt and hardware are installed properly and do not contact the governor body while moving the control arm through its full travel.

C. Rigging and Adjustment

(PIR-FTP2001-8, Rev. F., PIR-FTP2001-8-1, Rev. E. and PIR-FTP2007-1, Rev. C.)

(1) Start the engine; park the airplane 90° to the wind direction and warm in a normal manner until the oil temperature reaches 180°F.

(2) To check the high rpm, low pitch setting, move the propeller control all the way forward. At this position the governor speed control arm should be against the high rpm fine adjusting screw. With the throttle full forward, observe engine rpm, which should stabilize between 2,460 and 2,500 rpm. A takeoff must be conducted, during which the engine rpm should reach 2,460 - 2,500 rpm and remain steady.

(3) If the engine rpm does not read 2,460 - 2,500 rpm in flight, the high rpm setting must be adjusted as follows:

(a) Land and shut down the engine.

(b) Adjust the governor by means of the fine adjustment screw for 2,460 - 2,500 rpm. To do this, loosen the high rpm fine adjustment screw locknut and turn the screw in a clockwise direction to decrease engine speed or in a counterclockwise direction to increase engine speed.

NOTE: One revolution of the fine adjustment screw will increase or decrease the engine speed approximately 20 rpm.

(c) Repeat step (b), above, as required to obtain proper rpm setting.

(d) After setting the proper high rpm adjustment, run the self-locking nut on the fine adjustment screw against the base projection to lock.

(4) With the high rpm adjustment complete, the control system should be adjusted so that the governor control arm will contact the high rpm stop when the cockpit control knob is 0.032 to 0.047 of an inch from its full stop forward. Reconnect the cable end and tighten the jamnut. (Upper cowl must be removed to accomplish this adjustment.)

(5) It is usually only necessary to adjust the high rpm (low pitch) setting of the governor control system, as the action automatically takes care of the positive low rpm (high pitch) setting.

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CHAPTER

70

STANDARD PRACTICES - ENGINE

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CHAPTER 70

LIST OF EFFECTIVE PAGES

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CHAPTER 70 - STANDARD PRACTICES - ENGINES

TABLE OF CONTENTS

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GENERAL

Procedures

Use the following procedures whenever applicable:

A. To facilitate and ensure proper installation, tag and/or mark all parts and hardware as to their location before they are removed or disassembled.

B. When removing any tubes or engine parts, look for indications of scoring, burning or other undesirable conditions. Tag any unserviceable parts or units for investigation and possible repair.

C. Take extreme care to prevent foreign matter (lockwire, nuts, washers, dirt, etc.) from entering the engine whether it is on or off the aircraft. Make use of protective caps, plugs, and covers to ensure openings are not exposed.

CAUTION: DUST CAPS USED TO PROTECT OPEN LINES SHOULD ALWAYS BE INSTALLED OVER THE TUBE ENDS AND NOT IN THE TUBE ENDS. FLOW THROUGH THE LINES MAY BE BLOCKED OFF IF LINES ARE INADVERTENTLY INSTALLED WITH THE DUST CAPS IN THE TUBE ENDS

D. If anything is dropped into the engine, work should be stopped immediately and the item removed even if considerable time and labor is required.

E. Ensure all parts are thoroughly clean before assembling, especially during engine build-up.

F. Never reuse any lockwire, lock washers, tablocks, tab washers, or cotter pins.

G. All lockwire and cotter pins must fit snugly in holes drilled in specific hardware. On castellated nuts, the cotter pin head must fit into a recess of the nut with the other end bent such that one leg is back over the stud and the other is down flat against the nut. Use only corrosion resistant steel for cotter pins or lockwire.

H. When replacing gaskets, packings, or rubber parts, use the same type or composition as that of the gasket that was removed.

I. Make sure replacement nonmetallic parts show no sign of storage deterioration.

J. Use only a mallet of plastic or rawhide when installation of a part requires such force.

CAUTION: ENSURE THAT ANTI-SEIZE COMPOUNDS ARE APPLIED IN THIN EVEN COATS AND THAT EXCESS COMPOUND IS COMPLETELY REMOVED TO AVOID CONTAMINATION OF ADJACENT PARTS.

K. Loose-fitting spline drives external to the engine and having no means of lubrication should be lubricated with an anti-seize lubricant, such as molybdenum disulfide.

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CHAPTER

71

POWER PLANT

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CHAPTER 71

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
71-List of Effective Pages	1	Jun 25/20
	2	Sep 15/09
71-Table of Contents	1	Jun 25/20
	2	Sep 15/09
71-00-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Sep 15/09
	5	Sep 15/09
	6	Feb 15/14
	7	Sep 15/09
	8	Sep 15/09
	9	Jul 1/10
	10	Sep 15/09
	11	Sep 15/09
	12	Jul 1/10
	13	Jul 1/10
	14	Jul 1/10
	15	Jul 1/10
	16	Jul 1/10
	17	Jul 1/10
	18	Jul 1/10
	19	Jul 1/10
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	22	Jul 1/10
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	24	Jul 1/10
	25	Jul 1/10
	26	Jul 1/10
	27	Jul 1/10
	28	Jul 1/10
	29	Jul 1/10
	30	Jul 1/10
71-10-00	1	Sep 15/09
	2	Sep 15/09
	3	Sep 15/09
	4	Sep 15/09
71-20-00	1	Jun 25/20
	2	Jan 15/13
	3	Jan 15/13
	4	Feb 15/14
	5	Dec 15/14
	6	Dec 15/14
71-70-00	1	Sep 15/09
	2	Sep 15/09

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CHAPTER 71 - POWER PLANT

TABLE OF CONTENTS

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GENERAL

1. Description

The PA-46-350P, as well as the PA-46R-350T, is powered by a Lycoming TIO-540-AE2A series six-cylinder, direct-drive, wet-sump, horizontally-opposed, fuel-injected, dual-turbocharged, air-cooled engine with a compression ratio of 7.3:1, rated at 350 hp at 2,500 rpm and designed to operate on 100 or 100LL (minimum) octane aviation grade fuel.

The engine cowling is cantilevered from the firewall with secondary supports from the nose gear door frame. The engine cowling consists of an upper and dual lower sections which are fabricated from sheet metal with bonded stiffeners. The metal nose cowl is split through the vertical propeller centerline to permit complete cowl removal without removing the propeller.

The induction system consists of a dry-type air filter, an alternate air door, a Bendix servo regulator continuous flow type RSA-10ED1 fuel injector and an AN type fuel supply pump as an integral part of the fuel injector system. Dual AiResearch turbochargers are mounted as an integral part of the engine and incorporate bleed air sources for cabin pressurization (PA-46-350P only). Automatic waste gate control of the turbochargers provides constant air pressure at the fuel injector inlet, from sea level to critical altitude.

This engine is equipped with dual, pressurized, Slick type 6360 and 6363 magnetos. One magneto has a retard breaker providing fixed retard, long duration boosted spark for starting. A source of dc power and a starting vibrator are required to complete the installation. The spark advance is 20° BTC.

In addition to the aforementioned components, each engine is equipped with dual alternators, geared starter, and dual pneumatic vacuum pumps (if required). Engine mounts are steel tubing construction attached at the firewall and incorporate vibration absorbing dynafocal mounts. From the exhaust stacks, gases are directed to individual turbocharger exhaust plenums, through or around the turbo turbine, as required, and overboard at bottom rear of the engine cowling on each side.

The lubrication system is the pressurized wet-sump type. The oil pump, which is located in the accessory housing, draws oil through a drilled passage leading from the oil suction screen, located in the sump. The oil from the pump then enters a drilled passage in the accessory housing which feeds the oil to a threaded connection on the rear face of the accessory housing, where a flexible line leads the oil to the external oil cooler. Pressure oil from the oil cooler travels thru tubing to a remote pressure oil filter and then is returned thru tubing to a threaded connection on the accessory housing and into the engine. In the event that cold oil or an obstruction should restrict the oil flow through the cooler, an oil cooler bypass valve is provided to pass the oil directly from the oil pump to the oil pressure filter.

The oil filter element, located on the engine mount frame, provides a means of filtering from the oil any solid particles that may have passed through the suction screen in the sump. After being filtered, the oil is fed through a drilled passage to the oil pressure relief valve, located in the upper left side of the accessory housing at the rear of the engine. This relief valve regulates the engine oil pressure by allowing excessive oil to return to the sump, while the balance of the pressure oil is fed to the main oil gallery in the right half of the crankcase. Residual oil is returned by gravity to the sump, where after passing through a screen it is again circulated through the engine.

2. Troubleshooting

See “Chart 1”.

CHART 1 (Sheet 1 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Failure of engine to start.	Lack of fuel.	Check fuel system for leaks. Fill fuel tank. Clean dirty lines, strainers or fuel valves.
	Over priming.	Leave ignition OFF and mixture control in idle cut-off, open throttle and unload engine by cranking for a few seconds. Turn ignition switch on and proceed to start in a normal manner.
	Incorrect throttle setting.	Open throttle to one fourth of its range.
	Defective spark plugs.	Clean and adjust or replace spark plugs.
	Defective ignition wire.	Check with electric tester and replace any defective wires.
	Defective battery.	Replace with charged battery.
	Improper operation of magneto breaker.	Clean points. Check internal timing of magnetos.
	Lack of sufficient fuel flow.	Disconnect fuel line at fuel injector and check fuel flow.
	Water in fuel injector.	Drain fuel injector or fuel lines.
	Internal failure.	Check oil screens for metal particles. If found, complete overhaul of engine may be indicated.
Engine is hard starting.	Low voltage or defective vibrator.	Measure voltage between vibrator terminal marked IN and the ground terminal while operating starter. There must be at least 13 volts.

CHART 1 (Sheet 2 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Engine is hard starting. (continued)	Inoperative or defective vibrator.	If voltage is adequate, listen for buzzing of vibrator during starting. If no buzzing is heard, either the vibrator is defective or the circuit from the Output terminal on the vibrator to the retard contact assembly is open. Check both Switch and Retard circuits. Also check for good electrical ground.
	Retard contact assembly in magneto not operating electrically. Engine may kick back during cranking due to advance timing of ignition.	Retard points may not be closing due to wrong adjustment, or may not be electrically connected in circuit due to a poor connection. Inspect retard points to see if they close. Check for proper contact at the Switch and Retard leads at magneto and at the vibrator. Check wiring.
	Vibrator-magneto combination not putting out electrically.	Turn engine in proper direction of rotation until retard points just open No. 1 cylinder position. Remove input connection from starter to prevent engine turning and, while holding No. 1 plug lead 0.19 of an inch from ground, energize vibrator by activating the starter switch. Plug lead should throw a 0.19 of an inch spark. If spark is weak or missing, try new vibrator. If this does not correct trouble, check magneto for improper internal timing or improperly meshed distribution gears.
	Magneto improperly timed to engine.	Check magneto-to-engine timing.

CHART 1 (Sheet 3 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Engine is hard starting. (continued)	Advance contact assembly out of adjustment (internal timing off).	Check magneto timing.
	Retard points opening too late.	Check timing of retard points.
Failure of engine to idle properly.	Incorrect idle mixture.	Adjust mixture.
	Leak in the induction system.	Tighten all connections in the induction system. Replace any parts that are defective.
	Incorrect idle adjustment.	Adjust throttle stop to obtain correct idle.
	Uneven cylinder compression.	Check condition of piston rings and valve seats.
	Faulty ignition system.	Check entire ignition system.
	Insufficient fuel pressure.	Adjust fuel pressure.
	Leak in air bleed nozzle balance line.	Check connection and replace if necessary.
	Plugged fuel injector nozzle.	Clean or replace nozzle.
	Flow divider fitting plugged.	Clean fitting.
Low power and uneven running.	Mixture too rich as indicated by sluggish engine operation, red exhaust flame at night. Extreme cases indicated by black smoke from exhaust.	Readjustment of fuel injector by authorized personnel is indicated.
	Mixture too lean; indicated by overheating or backfiring.	Check fuel lines for dirt or other restrictions. Check fuel injection nozzles. Readjustment of fuel injector by authorized personnel is indicated.
	Leaks in induction system.	Tighten all connections. Replace defective parts.
	Defective spark plugs.	Clean and gap or replace spark plugs.
	Improper fuel.	Fill tank with fuel of recommended grade.

CHART 1 (Sheet 4 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Low power and uneven running. (continued)	Magneto breaker points not working properly.	Clean points. Check internal timing of magnetos.
	Defective ignition wire.	Check wire with electric tester. Replace defective wire.
	Defective spark plug terminal connectors.	Replace connectors on spark plug wire.
	Fuel injector nozzles clogged.	Clean or replace nozzle.
Failure of engine to develop full power.	Leak in the induction system.	Tighten all connections and replace defective parts.
	Throttle lever out of adjustment.	Adjust throttle lever.
	Improper fuel flow.	Check strainer, gauge and flow at fuel injector inlet.
	Restriction in air scoop.	Examine air scoop and remove restrictions.
	Improper fuel.	Drain and refill tank with recommended fuel.
	Plugged fuel injector nozzle.	Clean or replace nozzle.
	Propeller out of rig.	Adjust propeller.
	Faulty ignition.	Tighten all connections. Check system with tester. Check ignition timing.
Rough engine.	Cracked engine mount.	Repair or replace mount.
	Defective mounting bushings.	Install new mounting bushings.
	Uneven compression.	Check compression.
	Defective spark plugs.	Try new spark plugs.
	Defective plug leads.	Check plug leads for continuity and break down. Check distributor block for moisture and carbon tracking. Check contact springs in distributor block. Check magneto contact assemblies for burning or dirt (Main and Retard). Check distributor timing.

CHART 1 (Sheet 5 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Rough engine. (continued)	Magneto check out-of-limits.	Check magneto-to-engine timing. Inspect contact assemblies for proper opening.
	Blocked fuel injector nozzles.	Clean or replace nozzles.
Low oil pressure.	Insufficient oil.	Fill sump to proper level with recommended oil.
	Air lock or dirt in relief valve.	Remove and clean oil pressure relief valve.
	Leak in suction line or pressure line.	Check gasket between accessory housing and crankcase.
	Dirty oil strainers.	Remove and clean oil strainers.
	High oil temperature.	See High Oil Temperature in Trouble column, below.
	Defective pressure gauge.	Replace gauge.
	Stoppage in oil pump intake passage.	Check line for obstruction. Clean suction strainer.
High oil temperature.	Insufficient air cooling.	Check air inlet and outlet for deformation or obstruction.
	Insufficient oil supply.	Fill oil sump to proper level with specified oil.
	Low grade of oil.	Replace with oil conforming to specifications.
	Clogged oil lines, screens, or oil cooler.	Remove and clean oil screens. Replace or overhaul oil cooler.
	Excessive blow-by.	Usually caused by worn or stuck rings.
	Failing or failed bearings.	Examine sump for metal particles. If found, overhaul of engine is indicated.
	Defective temperature gauge.	Replace gauge.
	Defective thermostatic bypass valve.	Replace.

CHART 1 (Sheet 6 of 6)
TROUBLESHOOTING ENGINE

Trouble	Cause	Remedy
Excessive oil consumption.	Low grade of oil.	Fill sump with oil conforming to specifications.
	Failing or failed bearings.	Check sump for metal particles.
	Piston rings worn.	Install new rings.
	Incorrect installation of piston rings.	Install new rings.
	Failure of rings to seat. (New nitrided cylinders.)	Use mineral base oil. Climb to cruise altitude at full power and operate at 75% cruise power setting until oil consumption stabilizes.
High fuel flow indicated on fuel gauge.	Plugged fuel injector nozzle.	Clean or replace nozzle.

3. Engine

A. Removal

WARNING: PLACE A TAIL STAND UNDER THE TAIL OF THE AIRPLANE BEFORE REMOVING THE ENGINE.

NOTE: All hoses, lines, and wires should be tagged (identified) to facilitate installation. Open fuel, oil, and vacuum lines and fittings should be covered to prevent contamination.

(1) Turn off all electrical switches in the cockpit and then disconnect the battery ground wire at the battery.

(2) Move the fuel selector lever in the cockpit to OFF.

(3) Remove the engine cowling.

(4) Remove the propeller (refer to 61-10-00).

(5) Discharge the air conditioning system and disconnect the air conditioning compressor lines (refer to 21-50-00).

(6) Disconnect the starter positive and ground leads at the starter. Remove the lead attachment clamps.

(7) Disconnect the propeller governor control cable at the governor. Remove the cable attachment clamps.

(8) Disconnect the heated air hose from the heat muff.

(9) Disconnect the throttle and mixture cables at the injector.

(10) Remove the turbocharger air inlet ducts from both turbochargers.

(11) Disconnect the fuel pump supply line.

(12) Disconnect both engine oil lines.

(13) Disconnect the magneto P leads at the magnetos.

(14) Disconnect the engine vent tubes at the engine.

(15) Disconnect the engine oil temperature lead at the aft end of the engine.

(16) Disconnect the tachometer leads at the engine magneto sensors.

(17) Untie the ignition harness, hoses and lines at the aft end of the engine.

(18) Disconnect the vacuum pump lines at the pumps and remove the fittings from the pumps.

(19) Disconnect oil pressure line at the engine.

(20) Disconnect the manifold pressure line from the engine on the right side intake manifold.

(21) Disconnect the alternator leads and the cable attachment clamps.

(22) Attach a one-half ton (minimum) hoist to the hoisting straps and relieve the tension from the engine mounts.

(23) Check the engine for any attachments remaining to obstruct its removal.

(24) Drain the engine oil, if desired, and then close drain.

(25) Remove the four engine mount assemblies and swing the engine free, being careful not to damage any attaching parts.

(26) Place the engine on a suitable stand.

B. Installation

NOTE: Prior to installing the engine, install all items that were removed from the engine after the engine was removed from the aircraft. Remove any protective caps and identification tags as each item is installed.

(1) Attach a one-half ton (minimum) hoist to the engine and swing the engine into alignment with its attaching points.

NOTE: Refer to Figure 1 for the arrangement of the shock mount assemblies. The top shock mounts are assembled so the silver shocks are aft and the gold shocks are forward. The bottom shock mounts are installed opposite of the top shock mounts.

(2) Insert an engine mount bolt with washer against head into bottom motor mounts. Then insert heat shields, aft shock mount, and then spacer onto bolt.

(3) Lower engine against aft shock mount and add forward shock mount over bolt and spacer onto engine mount bracket.

(4) Place large wide washer then small washer over bolt, add nut and start only a few threads.

(5) Insert an engine mount bolt with washer against head into top motor mounts. Then insert aft shock mount and then spacer onto bolt.

(6) Raise engine against aft shock mount and add forward shock mount over bolt and spacer onto engine mount bracket.

(7) Place large wide washer then two small washers over bolt, add nut and start only a few threads.

(8) Tighten engine shock mounts (4) to torque specified in “Figure 1”.

(9) Tighten engine mount assembly bolts (5) to 240-270 inch pounds (torque bolt head).

(10) Connect alternator leads and secure the leads with the cable clamps.

(11) Connect the manifold pressure line to the right bottom side of the engine.

(12) Connect the oil pressure line.

(13) Install the fittings in the vacuum pumps (refer to 37-10-00) and attach the lines to the pumps.

(14) Connect the tachometer leads at the engine magneto sensors.

(15) Connect the engine oil temperature lead to the aft end of the engine.

(16) Connect engine vent tube to the engine.

(17) Connect the magneto P-leads to the magnetos.

(18) Connect oil filter lines to the oil cooler and engine.

(19) Connect the fuel pump supply line to the fuel pump.

(20) Attach the turbocharger air inlet ducts to the turbochargers.

(21) Connect the throttle and mixture cables to the injector.

(22) Connect the heated air hose to the heat muff.

(23) Connect the propeller governor control cable to the governor. Secure the control cable with the clamps.

(24) Connect the starter positive and ground leads to the starter. Secure the leads with the clamps.

(25) Connect the air conditioning compressor lines, if the air conditioning is installed.

(26) Ensure that any other lines, hoses or electrical leads that were disconnected during removal of the engine are now reconnected.

(27) Reinstall the propeller (refer to 61-10-00).

(28) Reinstall the engine cowling.

(29) Add the proper grade and amount of engine oil (refer to 12-10-00).

(30) Turn on the fuel valve, open the throttle full and turn on the electric fuel pump. Check the fuel lines for leaks.

(31) Service the air conditioning system.

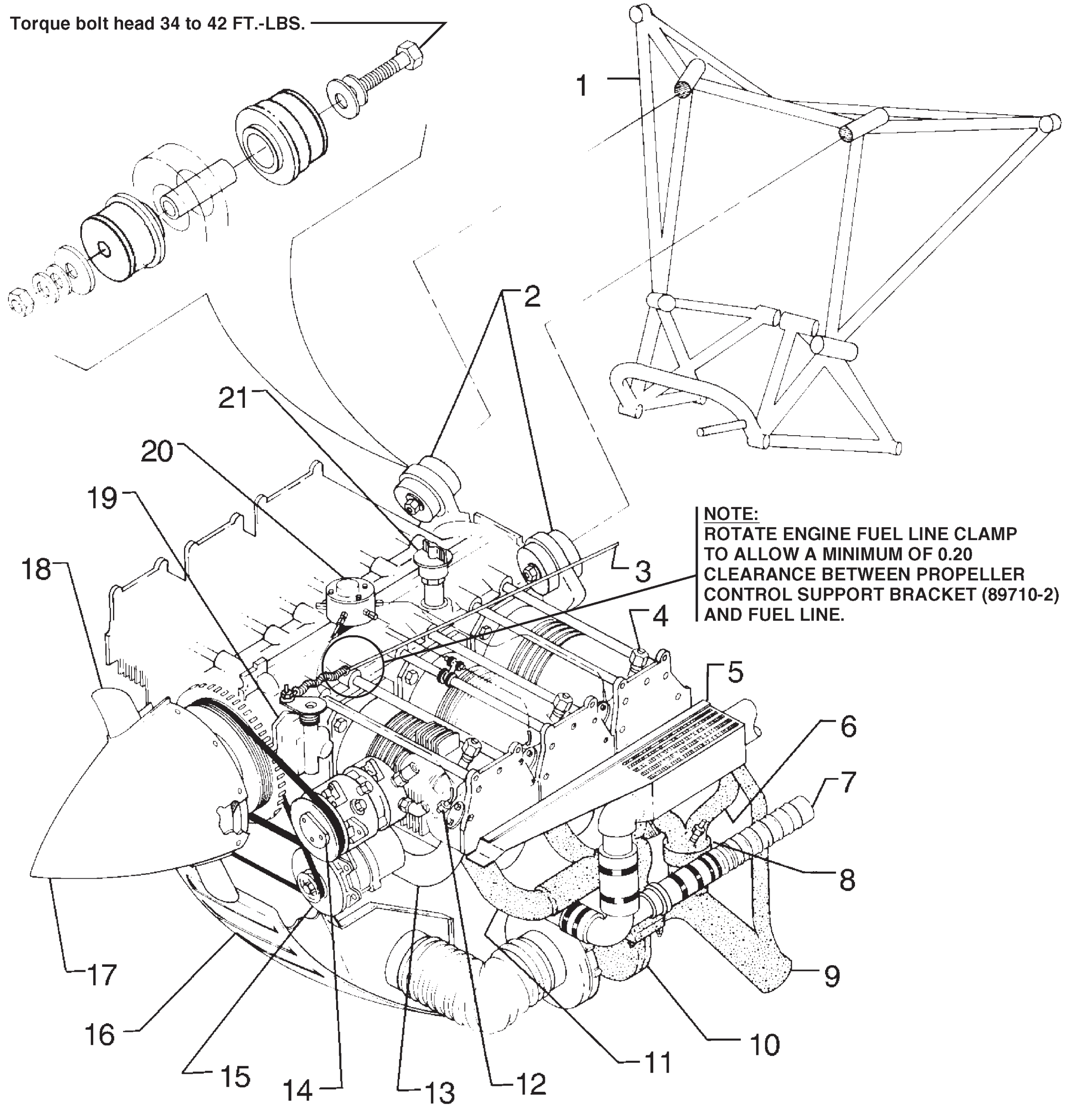
(32) Refer to 81-20-00 for information on pre-lubricating the turbochargers.

(33) Perform an engine operational check. Inspect for any leaks and make final adjustments to engine controls as required.

C. Component Locator

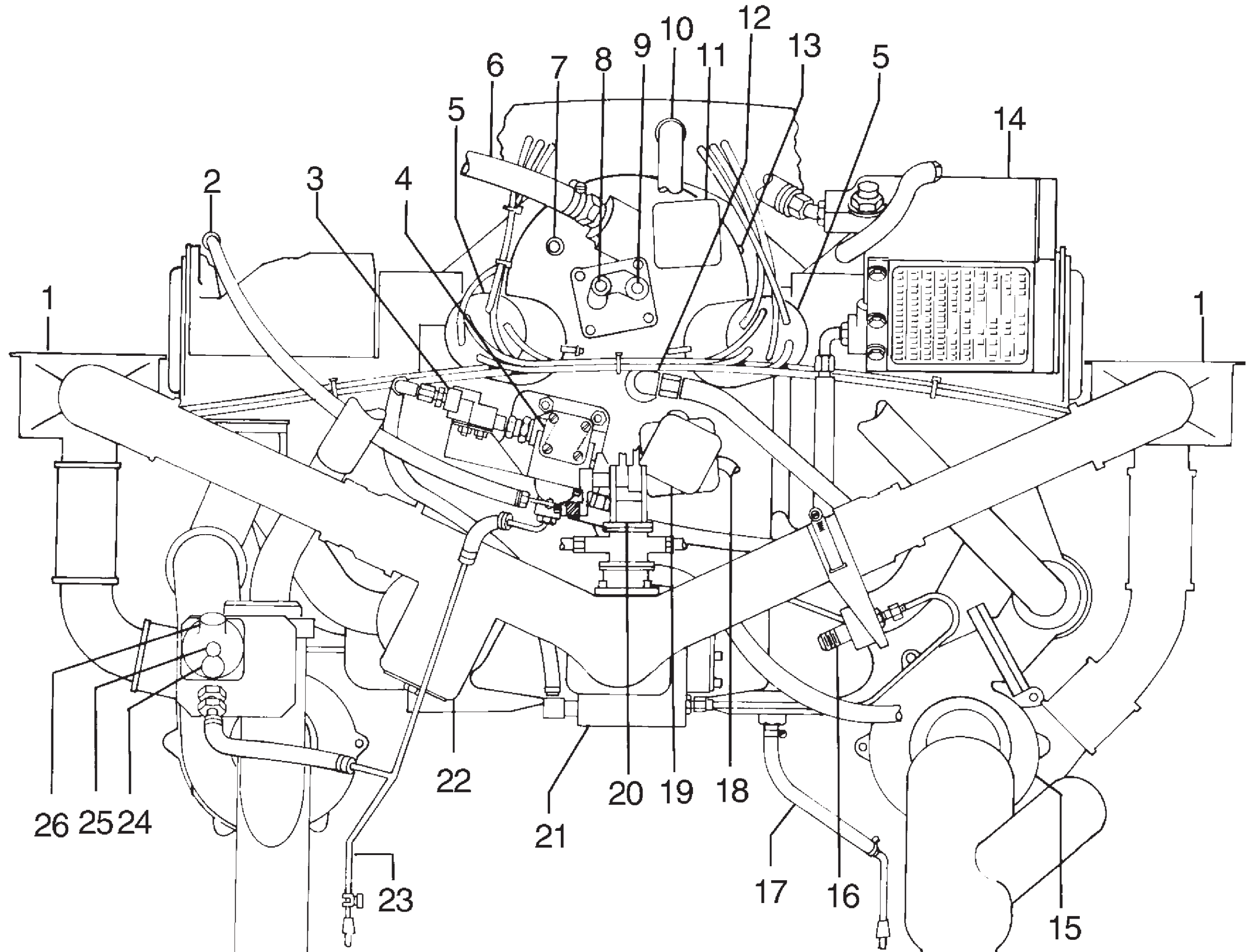
(PIR-PPS50079, Rev. A.)

See “Figure 2”.



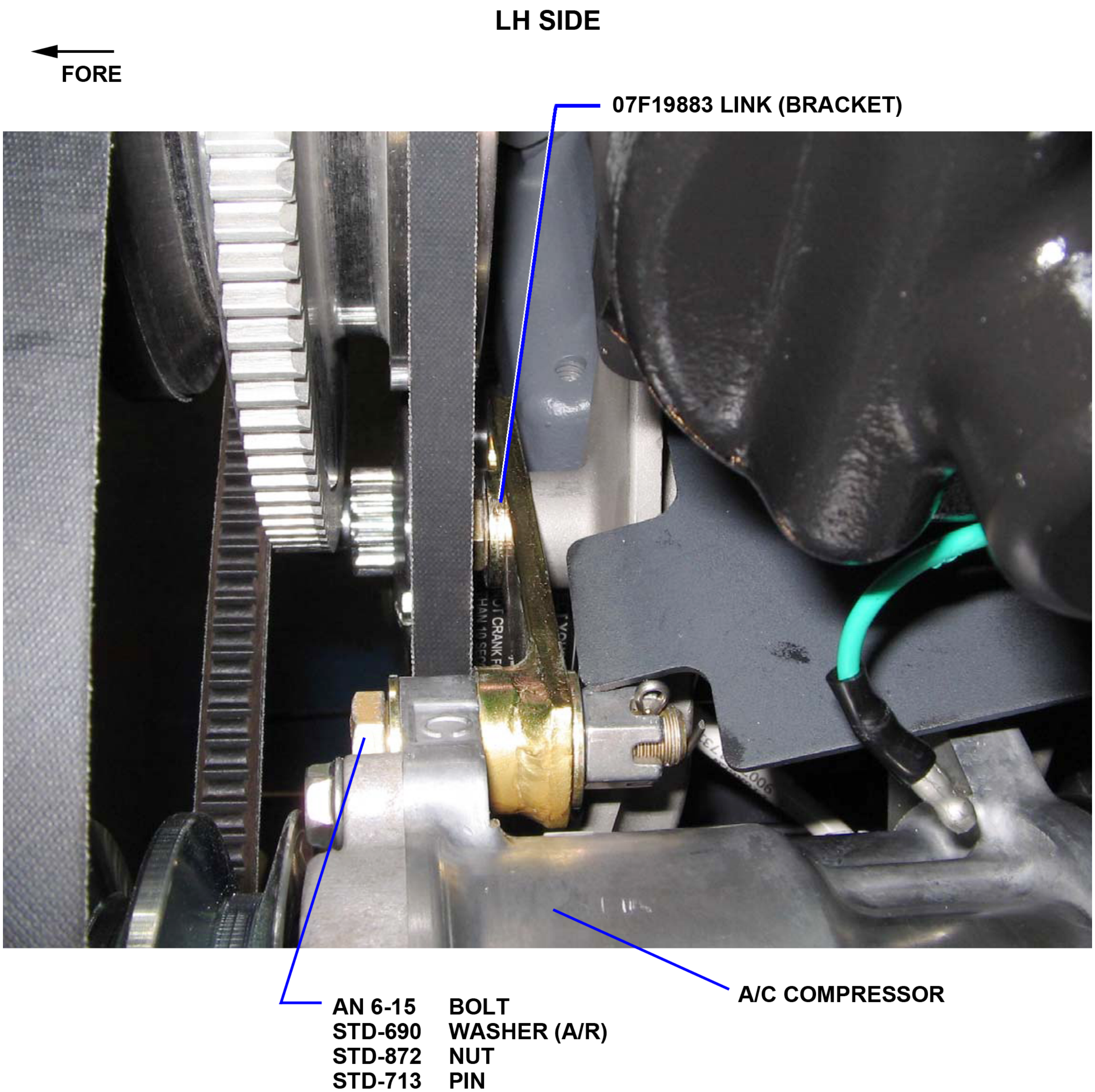
1. ENGINE MOUNT ASSEMBLY
2. ENGINE SHOCK MOUNTS (2 UPPER, 2 LOWER)
3. CONTROL CABLE TO PROPELLER GOVERNOR
4. SPARK PLUG
5. INTERCOOLER
6. EXHAUST MANIFOLD PRESSURE CROSSOVER
7. BLEED AIR PICKUP FOR PRESSURIZATION
8. WASTEGATE
9. EXHAUST PIPE
10. TURBOCHARGER
11. EXHAUST MANIFOLD
12. FUEL / AIR INJECTION NOZZLE
13. INTAKE MANIFOLD
14. AIR CONDITIONING COMPRESSOR
15. ALTERNATOR
16. AIR BOX
17. SPINNER
18. PROPELLER
19. PROPELLER GOVERNOR
20. FUEL FLOW DIVIDER
21. OIL DIPSTICK AND FILLER NECK

Engine Installation
Figure 1 (Sheet 1 of 2)



1. INTERCOOLER
2. DECK PRESSURE FOR FUEL PUMP & NOZZLES
3. "AN" FUEL PUMP
4. FUEL FLOW TRANSDUCER
5. MAGNETO CONTROLLER
6. OIL BREATHER TUBE TO OIL SEPARATOR
7. OIL RETURN FROM FILTER
8. OIL PRESSURE TO LEFT TURBOCHARGER
9. THERMOCOUPLE PORT
10. FUEL LINE FROM FUEL INJECTOR SERVO TO FUEL FLOW DIVIDER DRAIN
11. VACUUM PUMP PAD #1
12. OIL TO COOLER CONTROLLER
13. TO OIL PRESSURE GAUGE
14. OIL COOLER
15. TURBOCHARGER
16. FUEL PRESSURE SWITCH
17. FUEL DRAIN
18. OIL RETURN FROM VARIABLE PRESSURE
19. VACUUM PUMP PAD #2
20. VARIABLE PRESSURE CONTROLLER
21. FUEL INJECTOR SERVO
22. ABSOLUTE PRESSURE CONTROLLER
23. FUEL PUMP RELIEF AND WASTEGATE
24. OIL PRESSURE TO VARIABLE PRESSURE
25. WASTEGATE CONTROLLER
26. ENGINE OIL TO WASTEGATE CONTROLLER

Engine Installation
Figure 1 (Sheet 2 of 2)

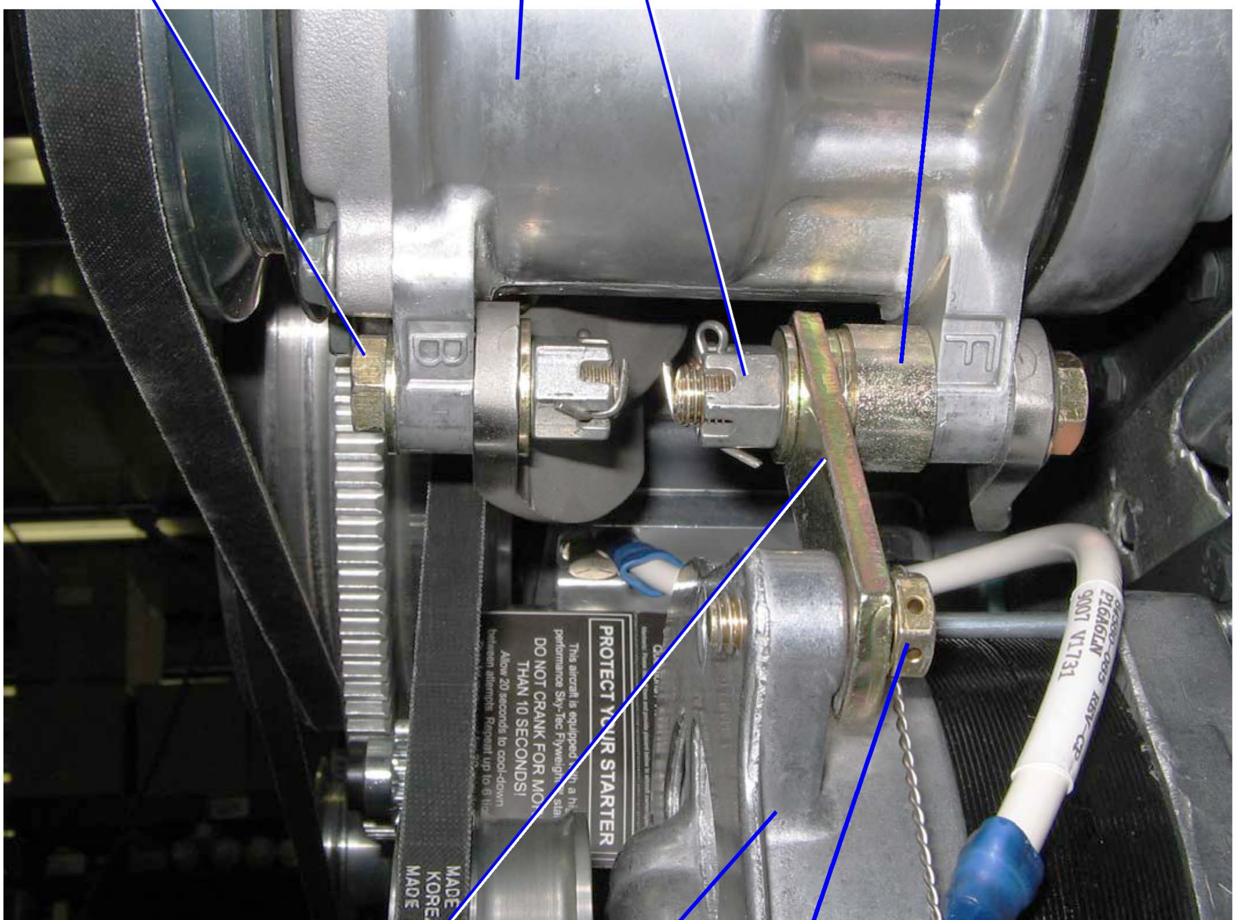


NOTE: Unless standard parts or otherwise noted, all parts numbers in this figure are Lycoming.

Engine Component Locator
Figure 2 (Sheet 1 of 19)

LH SIDE (CLOSE UP)

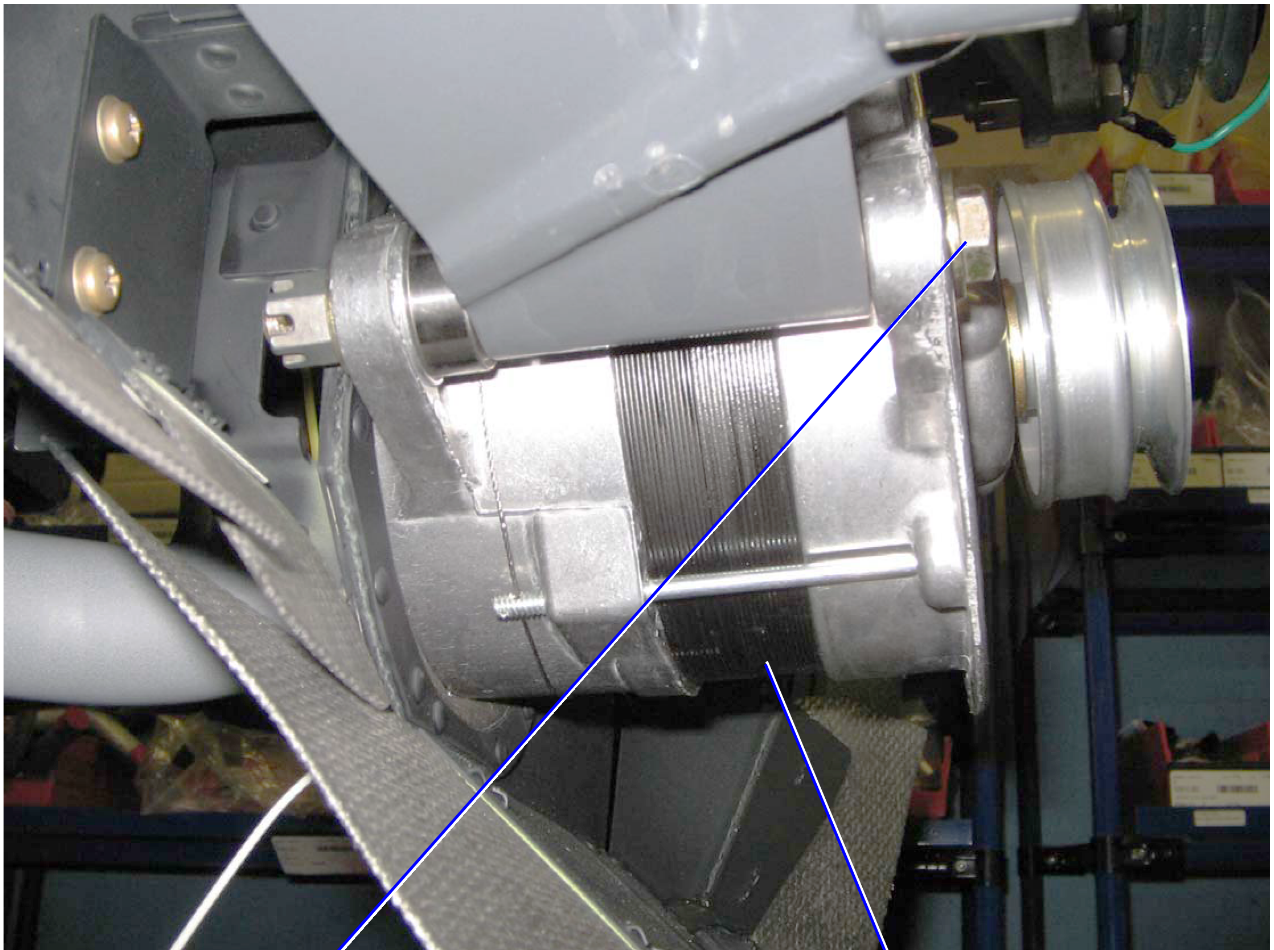
Looking to RH Side



NOTE: See Note in Sheet 1 of this figure.

Engine Component Locator
Figure 2 (Sheet 2 of 19)

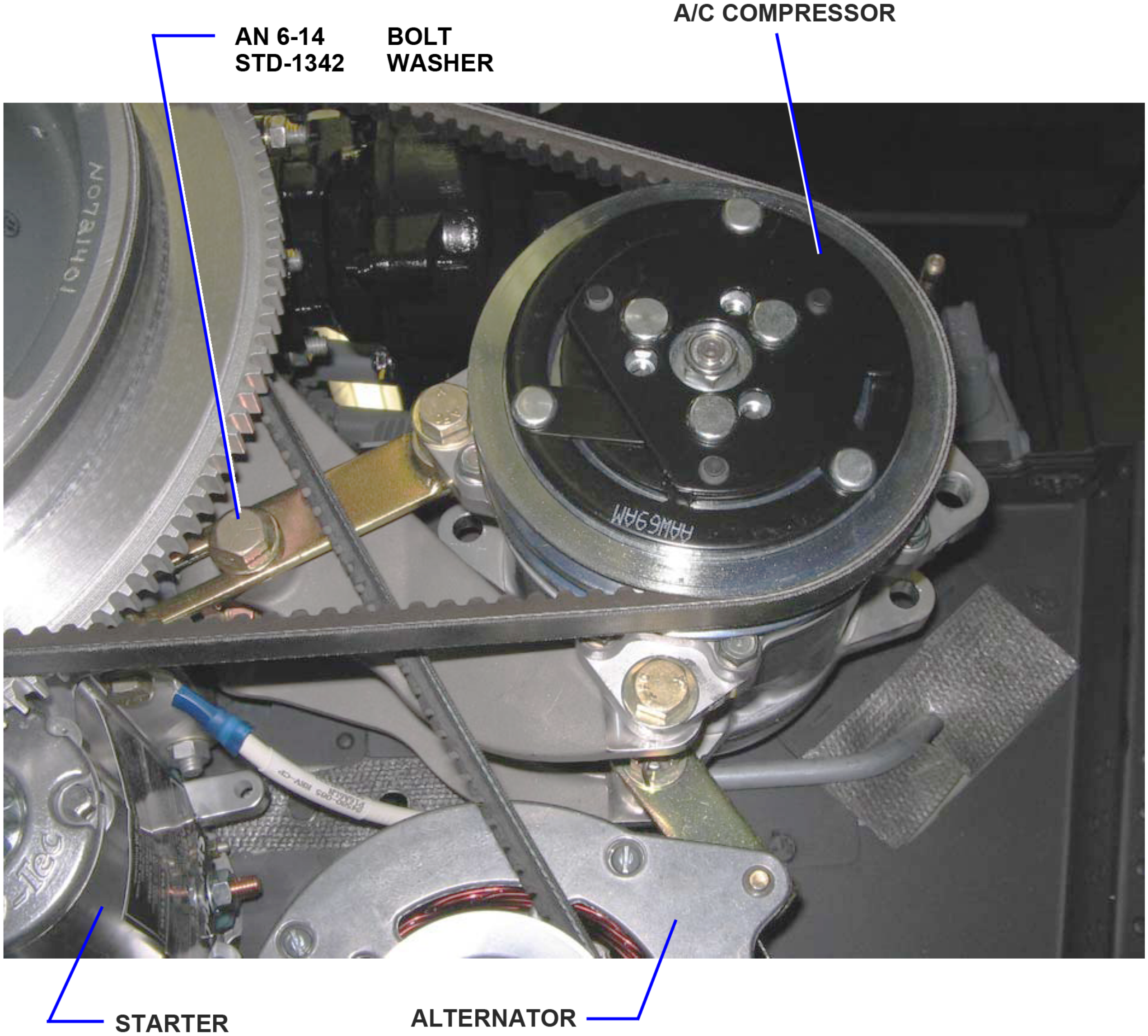
RH SIDE
Looking from Below, to LH Side



NOTE: See Note in Sheet 1 of this figure.

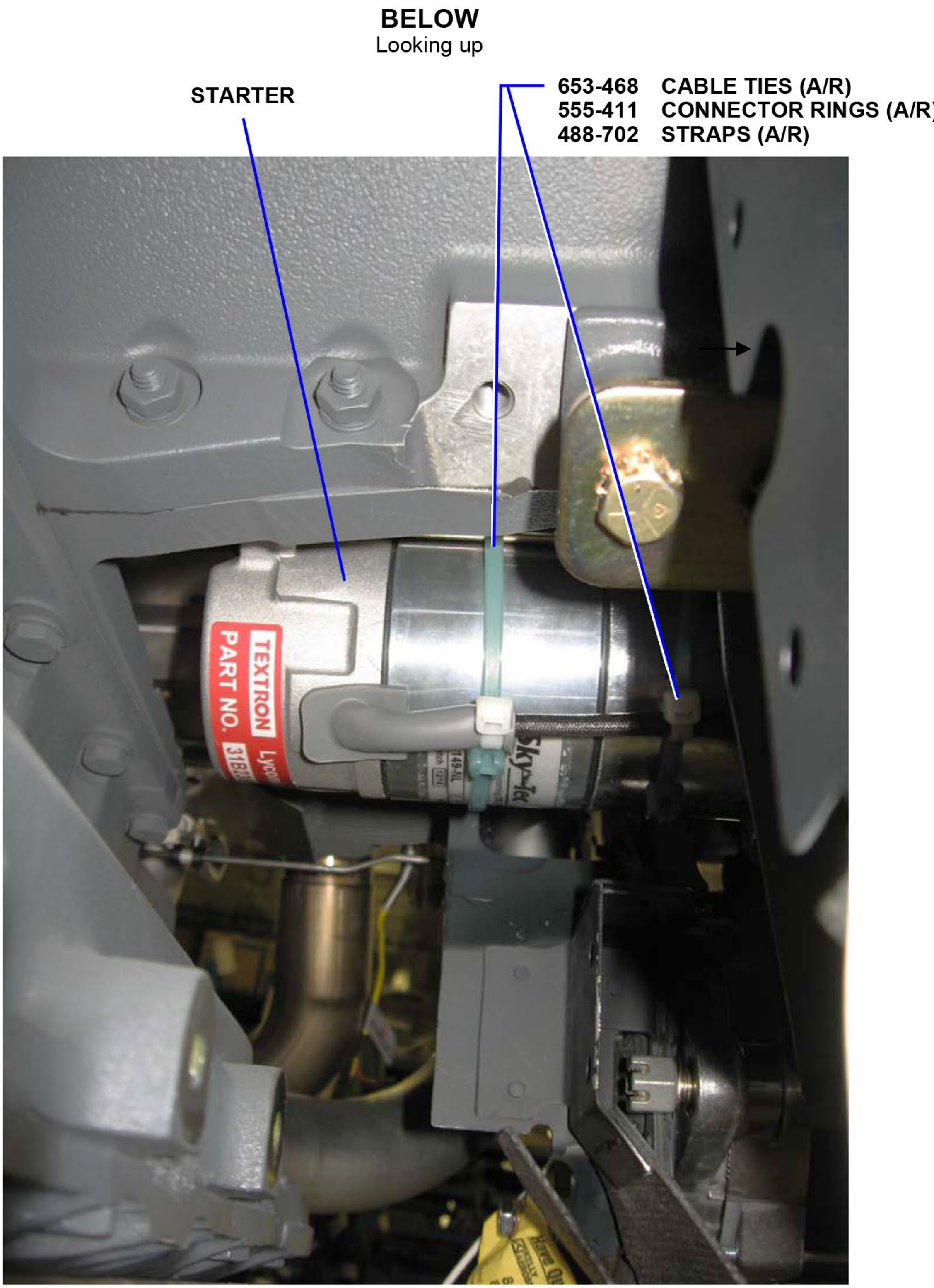
Engine Component Locator
Figure 2 (Sheet 3 of 19)

FRONT
From Below, Looking Aft



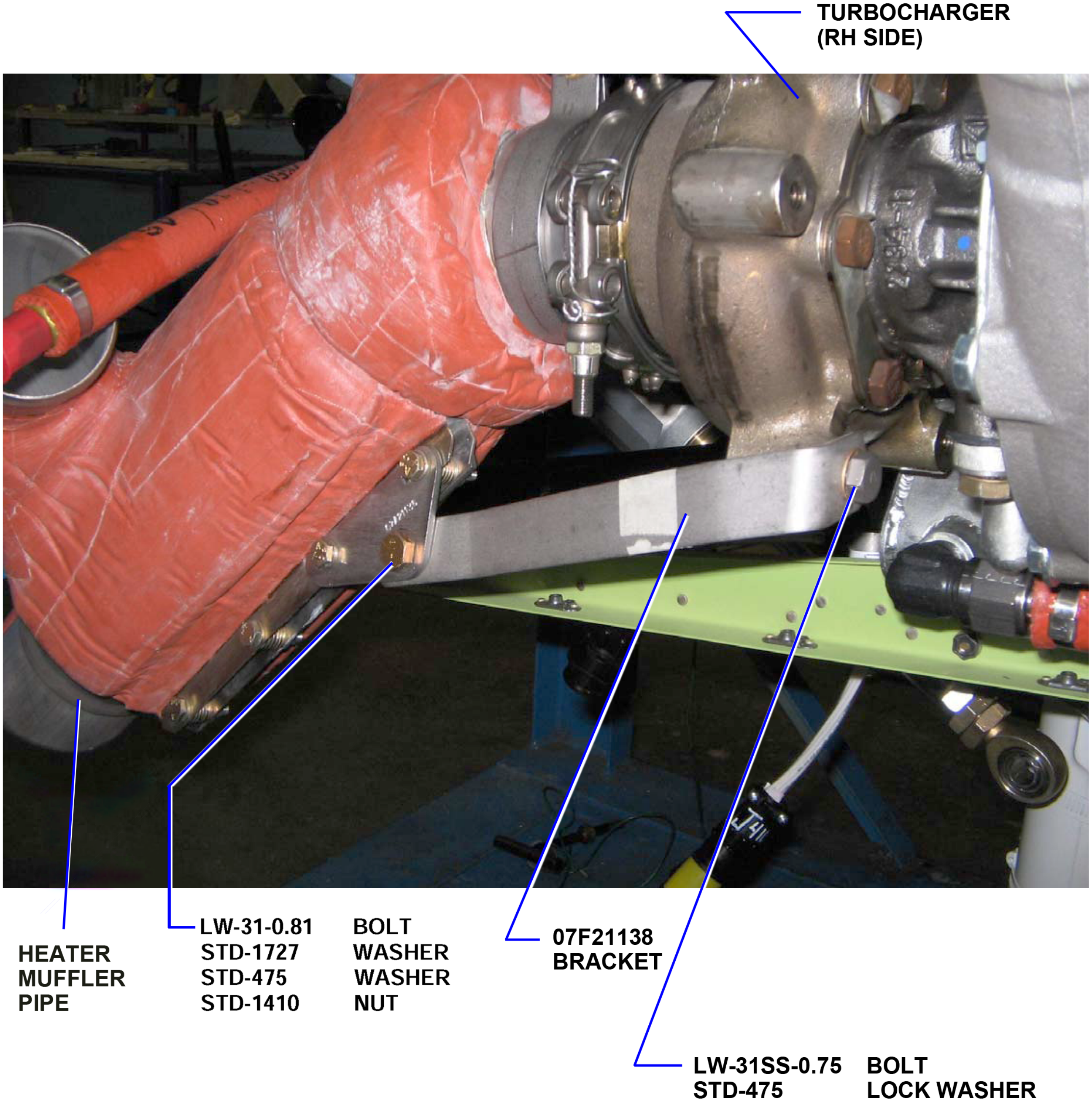
NOTE: See Note in Sheet 1 of this figure.

Engine Component Locator
Figure 2 (Sheet 4 of 19)



NOTE: See Note in Sheet 1 of this figure.

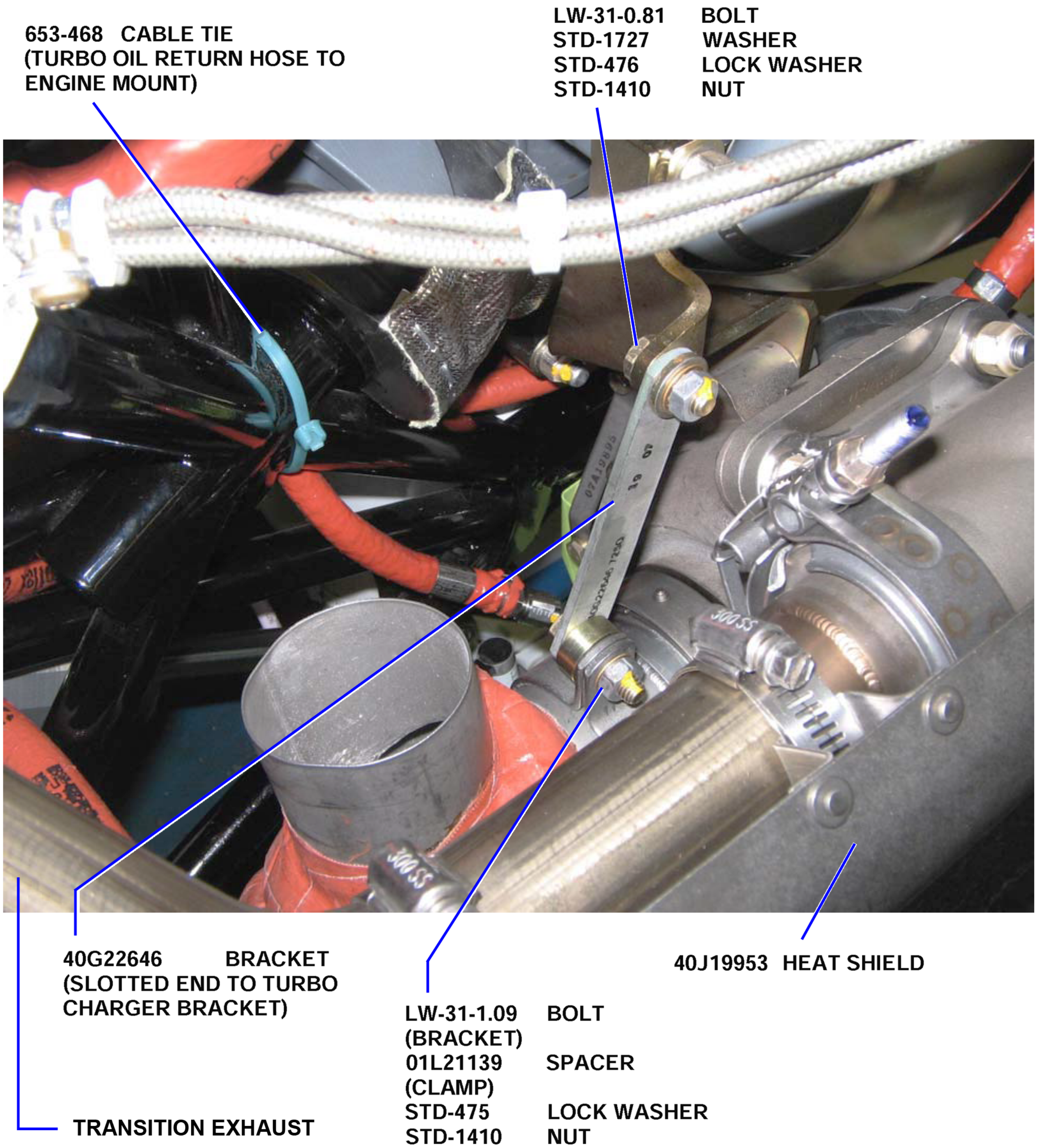
Engine Component Locator
Figure 2 (Sheet 5 of 19)



NOTE: See Note in Sheet 1 of this figure.

RH SIDE
Looking Left

Engine Component Locator
Figure 2 (Sheet 6 of 19)

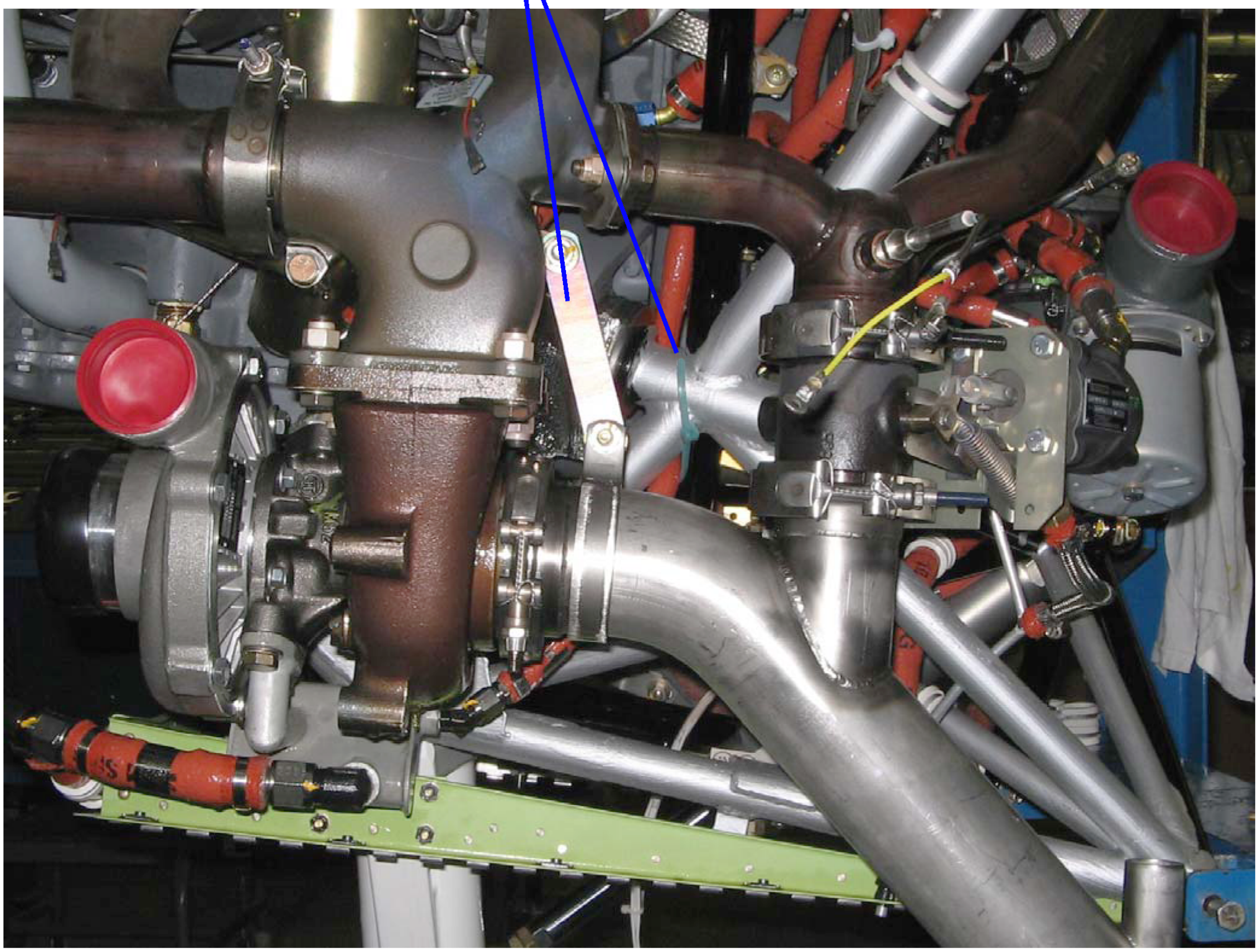


NOTE: See Note in Sheet 1 of this figure.

RH SIDE (CLOSE UP)
(LH SIDE SIMILAR)

Engine Component Locator
Figure 2 (Sheet 7 of 19)

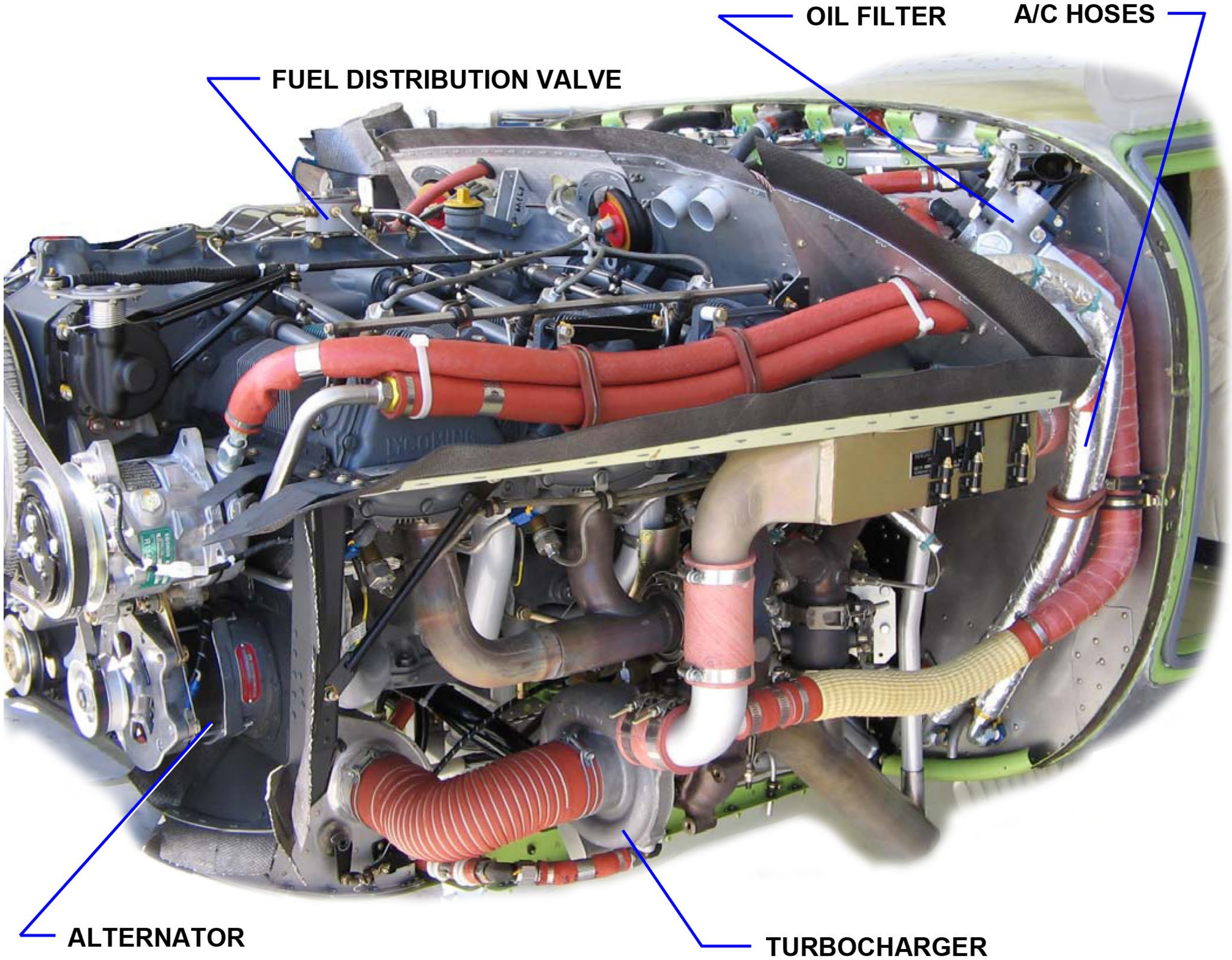
SEE SHEET 7, “RIGHT SIDE (CLOSE UP)” VIEW



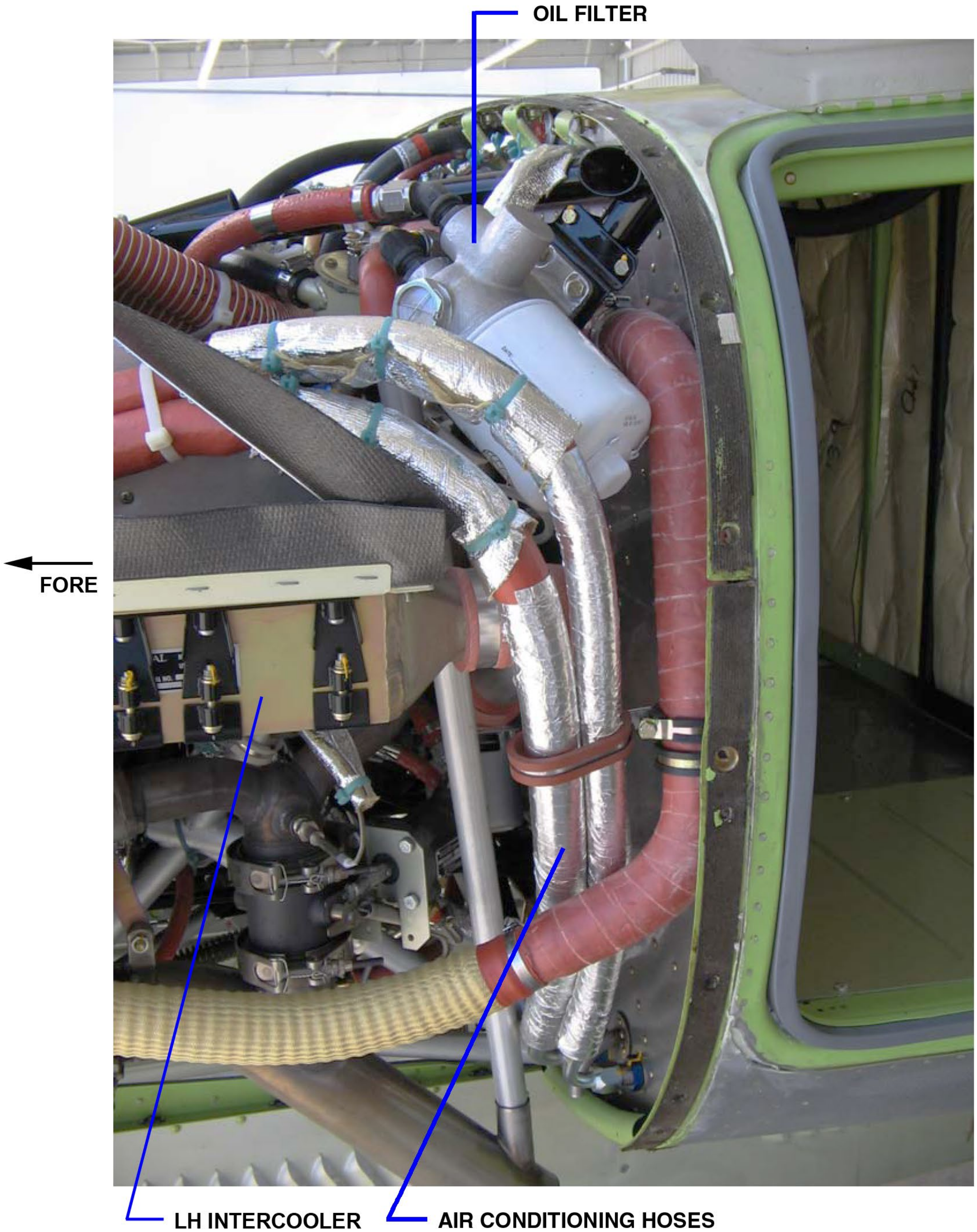
LEFT SIDE

Engine Component Locator
Figure 2 (Sheet 8 of 19)

FWD LEFT
Looking Aft Right from LH Side

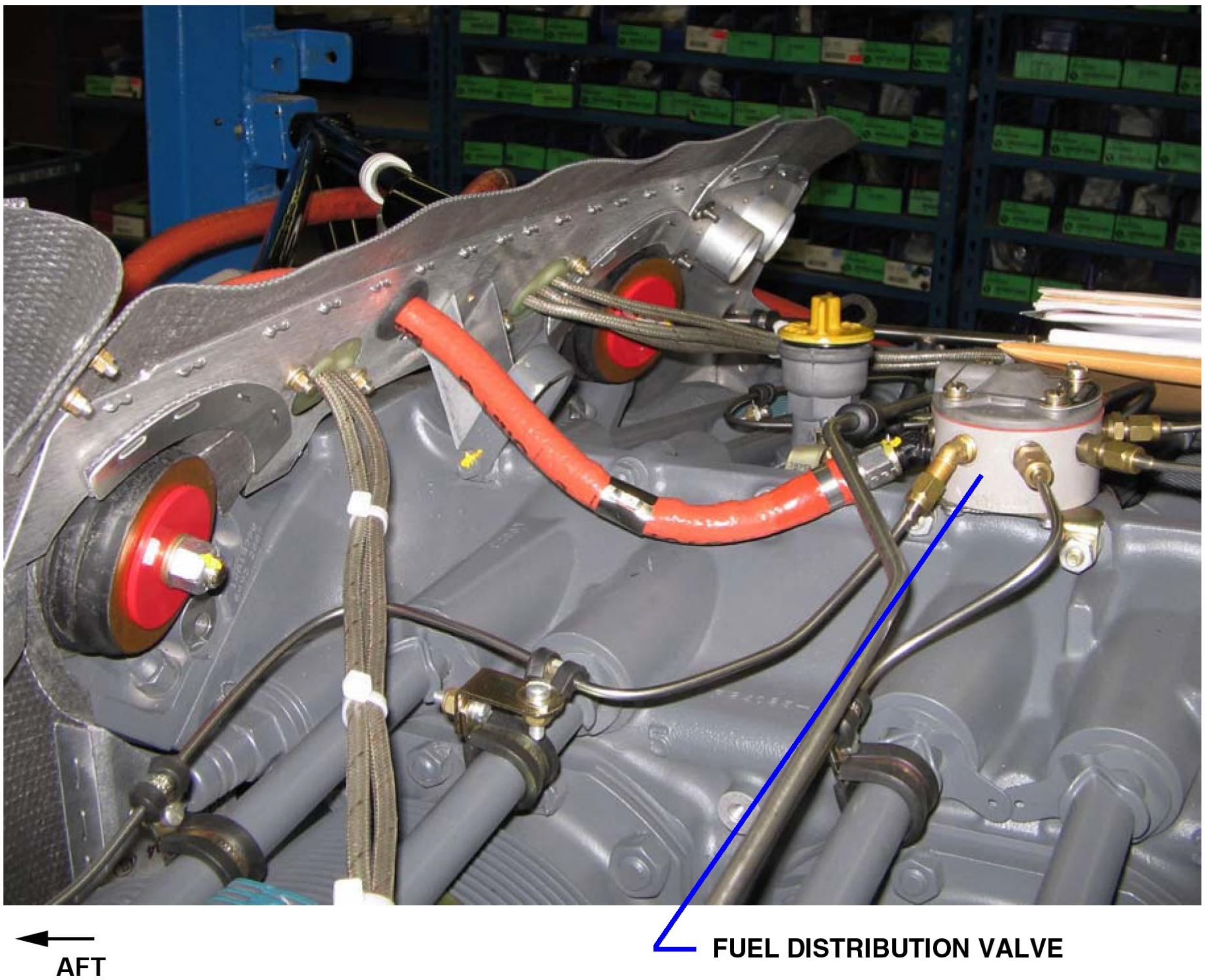


Engine Component Locator
Figure 2 (Sheet 9 of 19)

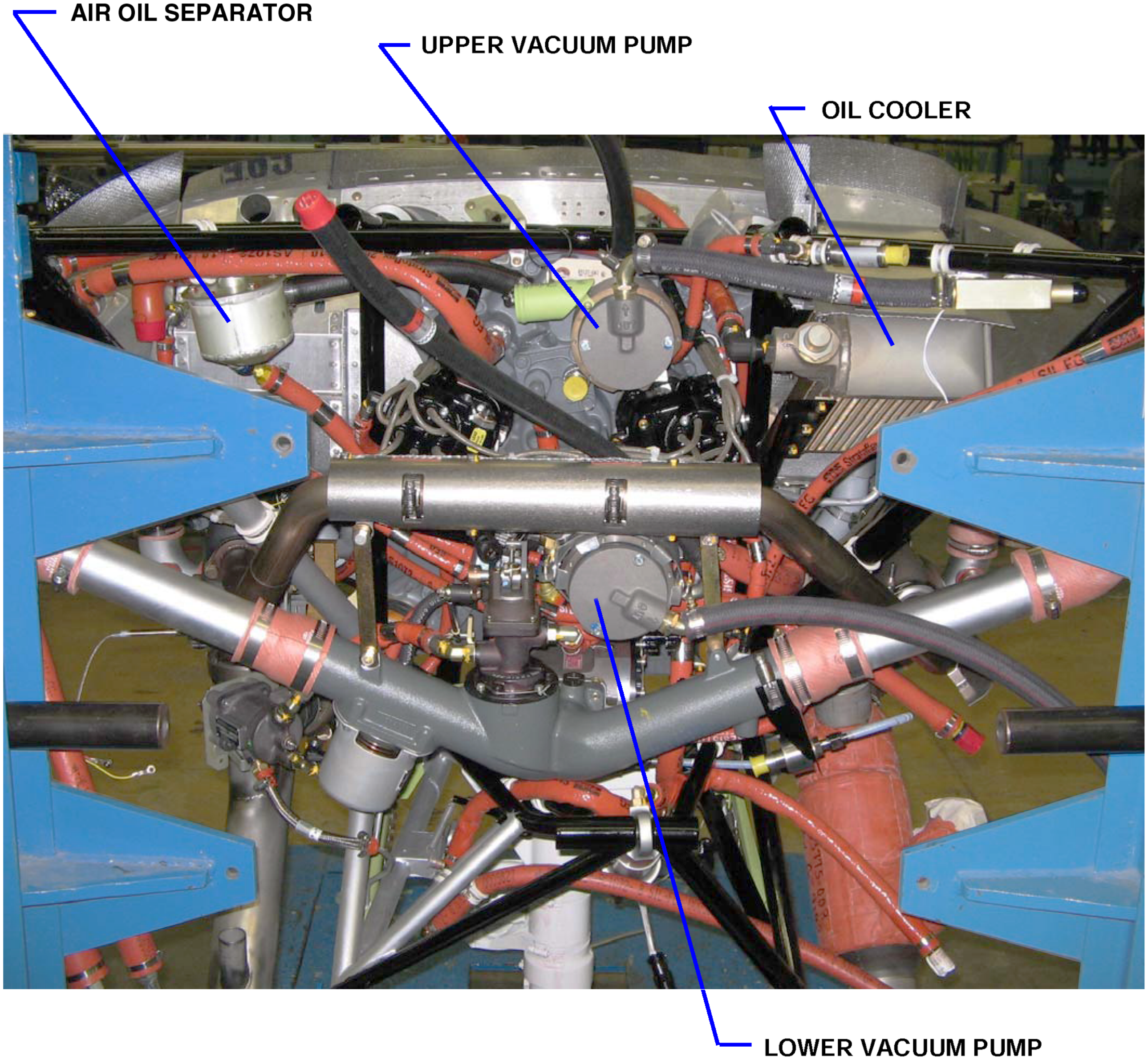


Engine Component Locator
Figure 2 (Sheet 10 of 19)

RH SIDE
Looking Down to LH Side

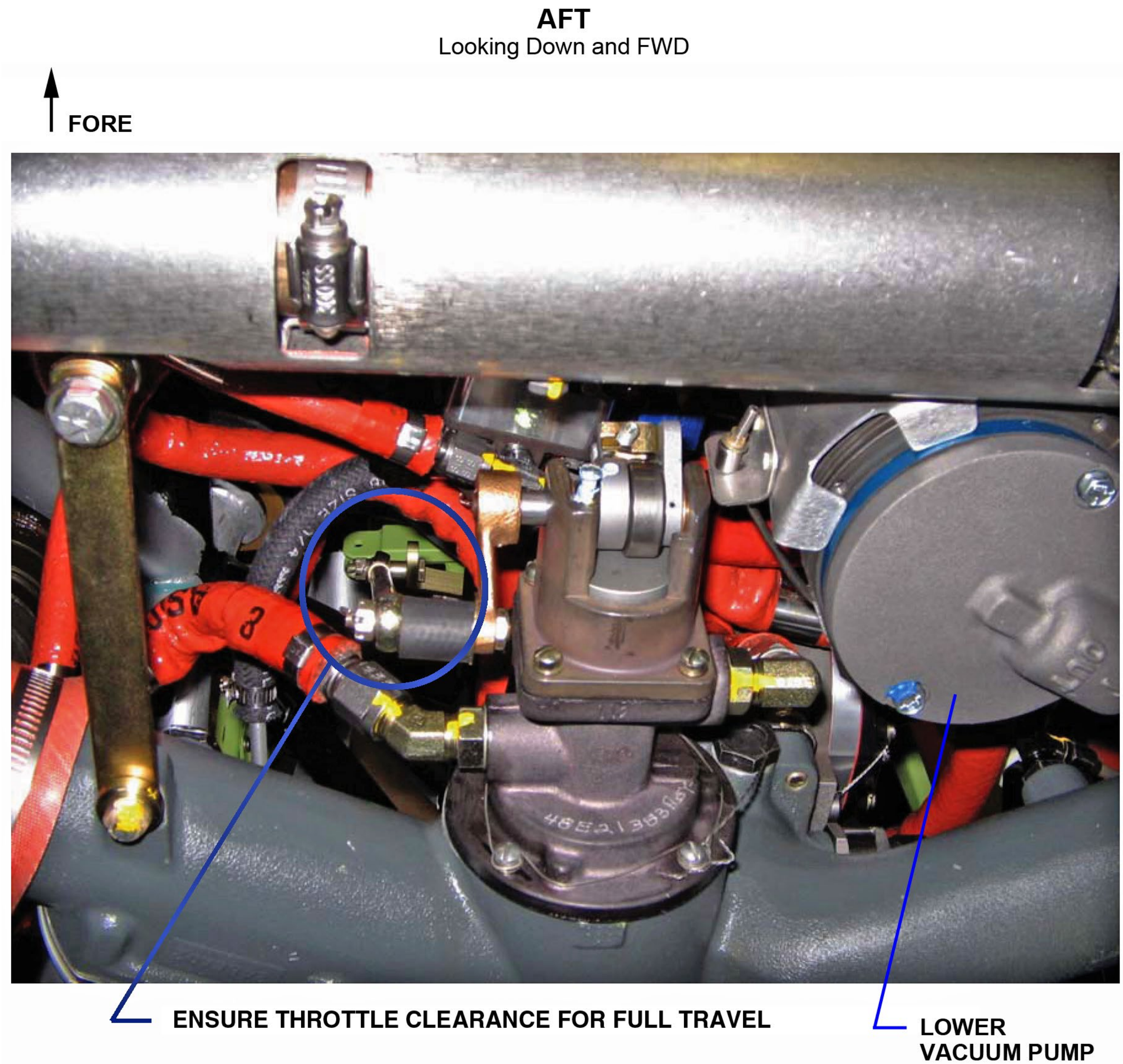


Engine Component Locator
Figure 2 (Sheet 11 of 19)



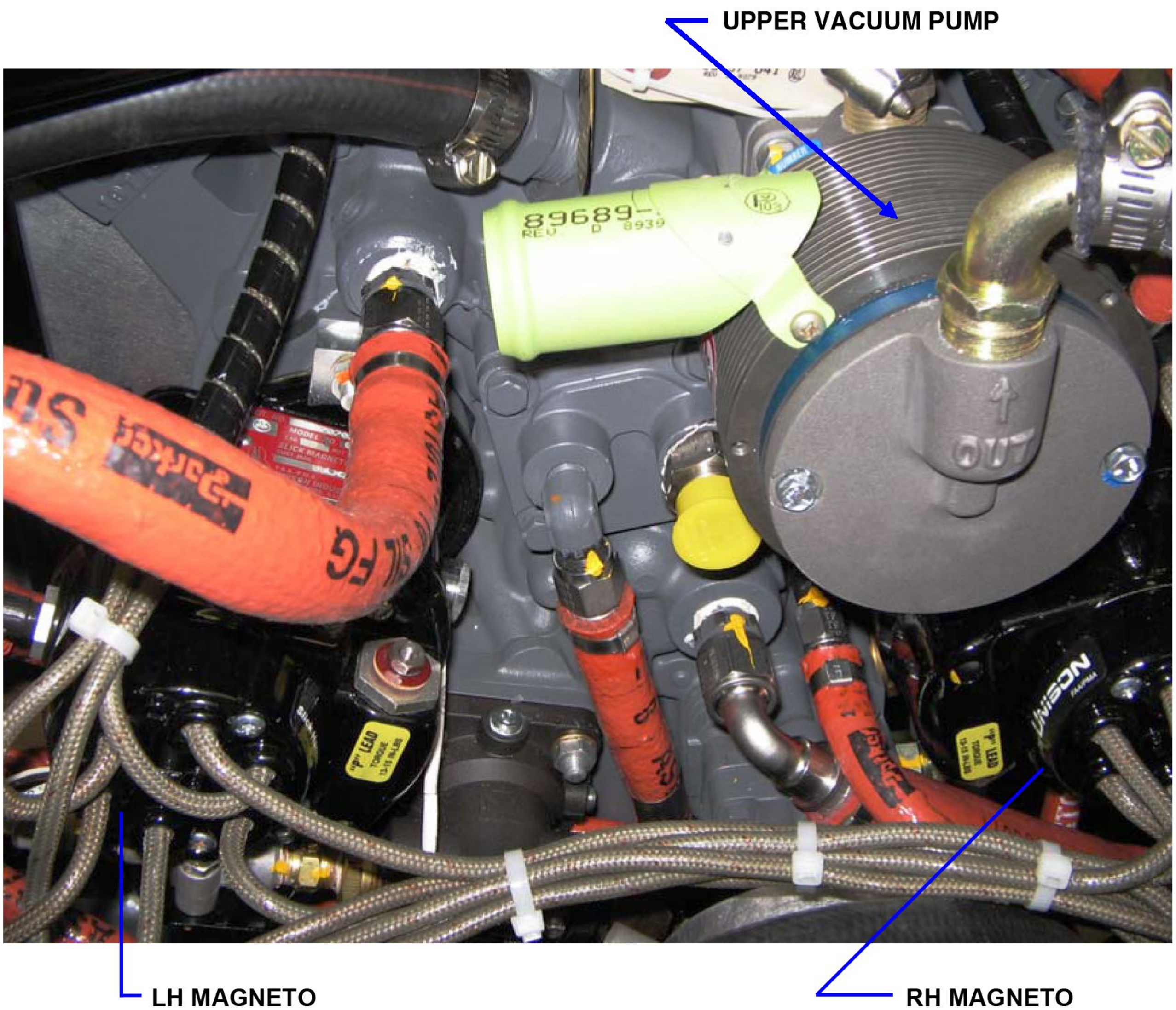
AFT
Looking FWD

Engine Component Locator
Figure 2 (Sheet 12 of 19)

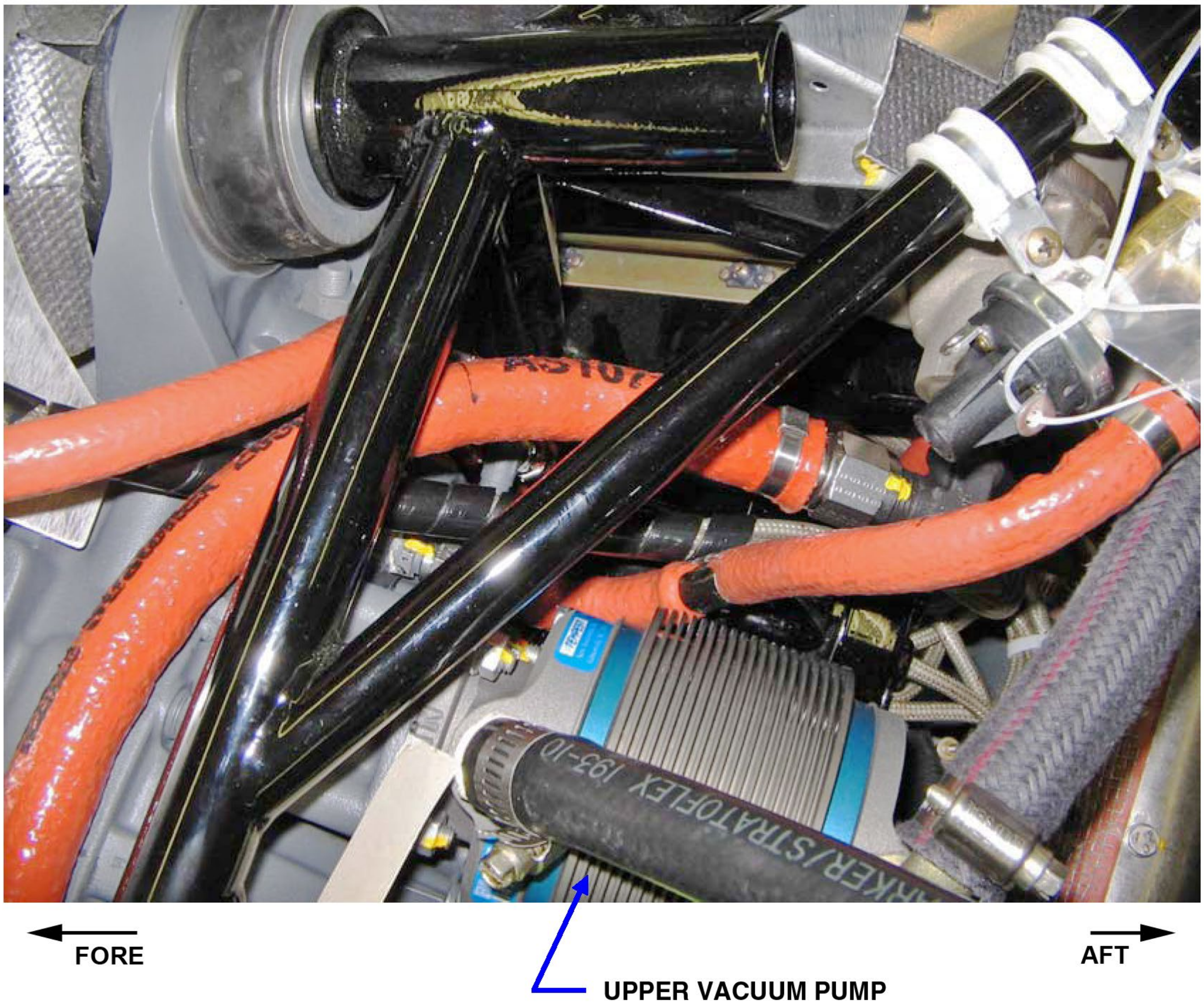


Engine Component Locator
Figure 2 (Sheet 13 of 19)

AFT
Looking FWD

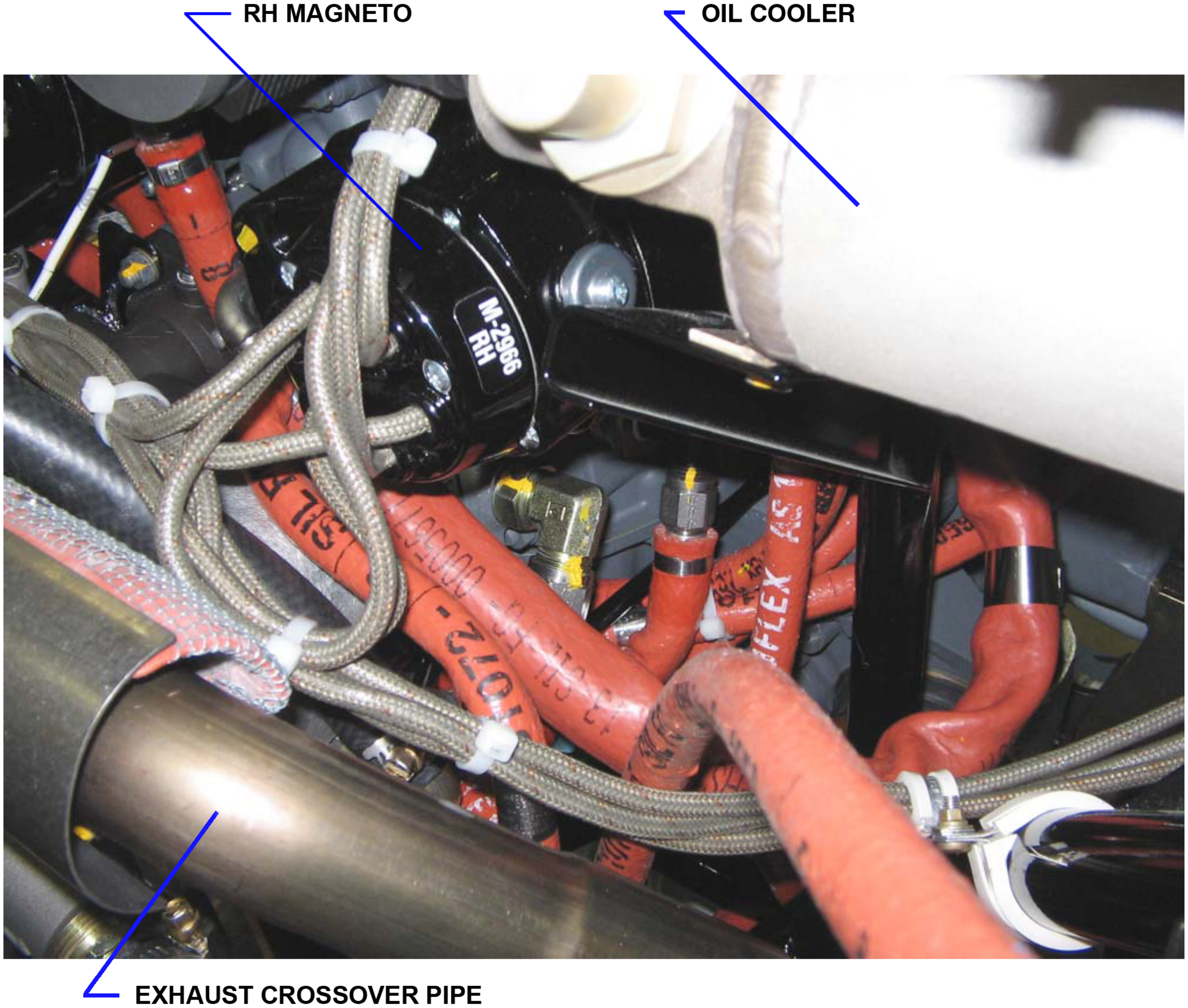


Engine Component Locator
Figure 2 (Sheet 14 of 19)



RH SIDE
View Looking Down from LH Side

Engine Component Locator
Figure 2 (Sheet 15 of 19)

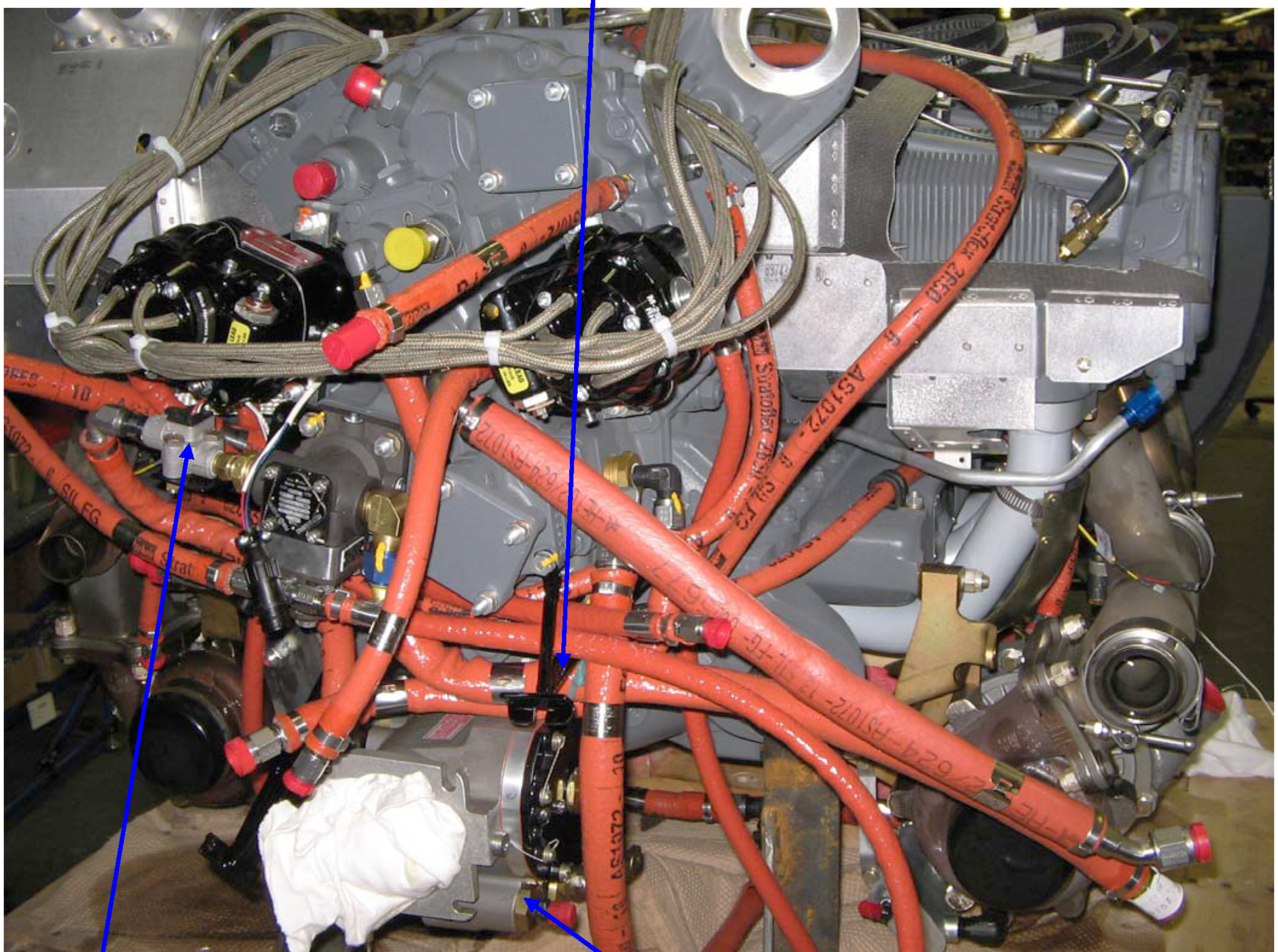


AFT RH SIDE
Looking Down toward FWD Left
(Off Airframe)

Engine Component Locator
Figure 2 (Sheet 16 of 19)

AFT, RH SIDE
Looking FWD

653-468 CABLE TIE
(FUEL HOSE TO MIXTURE BRACKET)



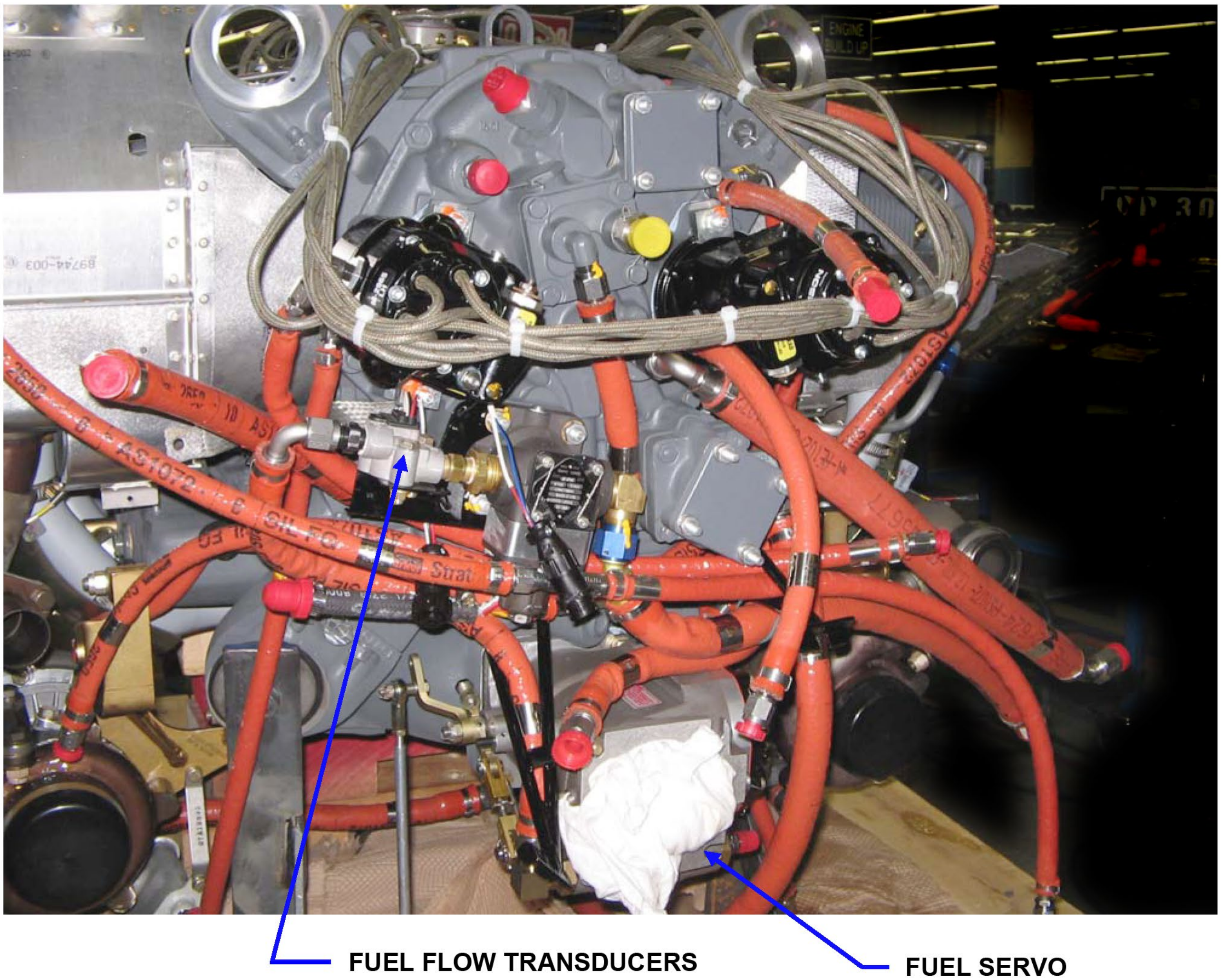
FUEL FLOW TRANSDUCER

FUEL SERVO

NOTE: See Note in Sheet 1 of this figure.

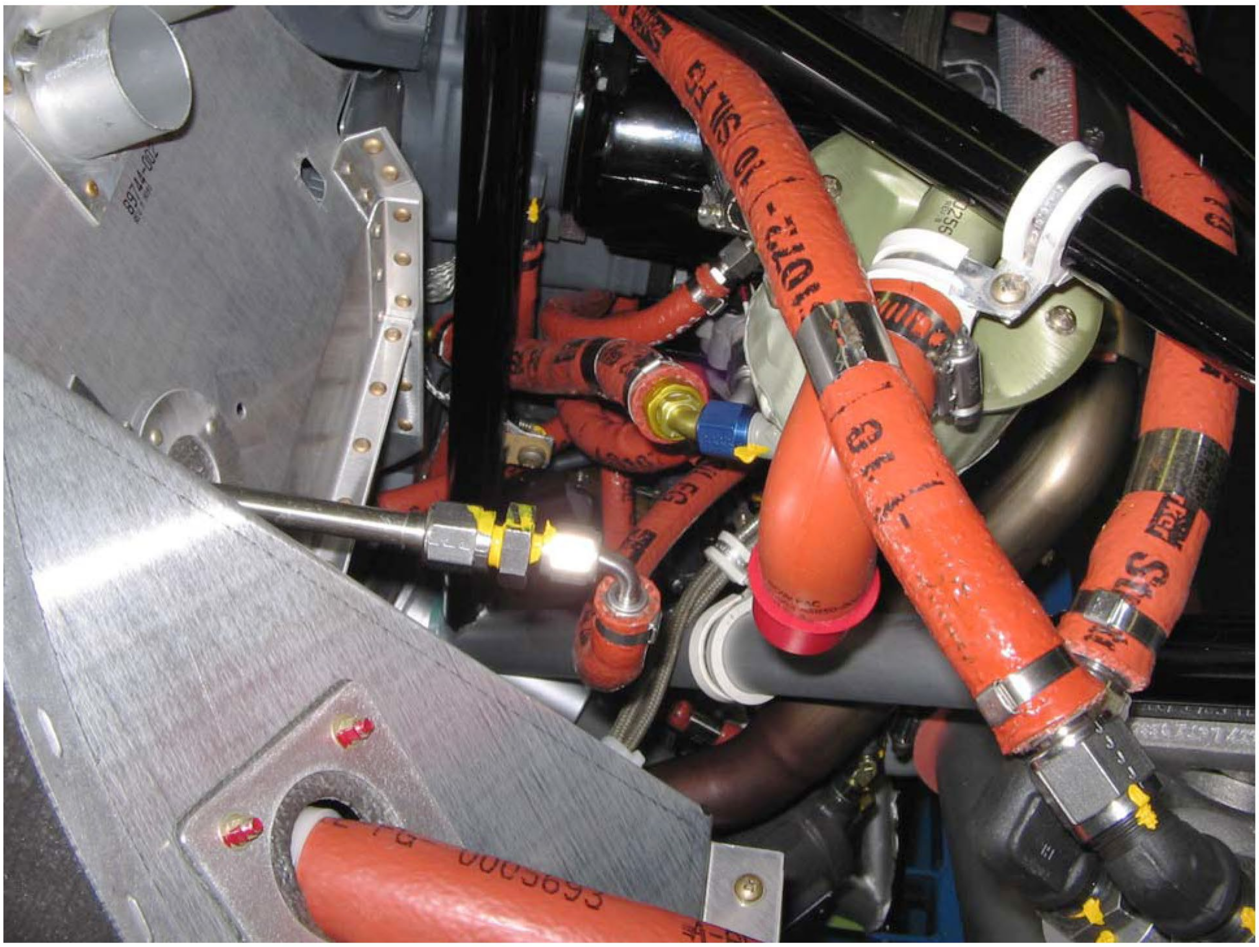
Engine Component Locator
Figure 2 (Sheet 17 of 19)

AFT
Looking Forward



Engine Component Locator
Figure 2 (Sheet 18 of 19)

FORE



LEFT SIDE
Looking down from LH (Pilot) Side
toward RH FWD

Engine Component Locator
Figure 2 (Sheet 19 of 19)

COWLING

Engine Cowling

(See “Figure 1”)

A. Removal

(1) Upper

(a) Release the fasteners that attach the upper cowl to the nose cowl and lower cowls.

(b) Release the fasteners that attach the upper cowl to the firewall flange.

(c) Remove the upper cowl.

(2) Lower (left and right)

NOTE: Support left and right lower cowls before removing any attaching hardware.

(a) Release clamp attaching heater muff hose to the NACA inlet on right cowl.

(b) Release clamp attaching hose from NACA inlet on right cowl to air modulator box.

(c) Remove screws that attach the lower cowl to the nose cowl.

(d) Remove the screws that attach the lower inboard edges of the lower cowl to the induction air panel assembly and along nose gear doors.

(e) Remove the screws that attach the lower cowl to the firewall flange.

(f) Remove each lower cowl.

(3) Nose Bowl

(a) Remove the screws that attach the lower inboard edges of the nose cowl to the induction air panel assembly.

(b) Remove the five upper and two lower screws and washers attaching the right half of the nose cowl to the left half.

(c) Remove nose bowl cowl.

B. Installation

(1) Place the nose cowl halves around the propeller drive shaft and secure them to each other with the five upper and two lower screws and washers.

(2) Position both halves of the lower cowling in place and secure them to the nose cowl, firewall flange and along nose gear doors with screws.

(3) Install screws attaching induction air panel assembly to the lower inboard edges of the lower cowls.

(4) Install the screws that attach the lower inboard edges of the nose cowl to the induction air panel assembly.

(5) Attach hose from NACA inlet on right cowl to air modulator box. Tighten clamp.

(6) Attach heater muff hose to the NACA inlet on right cowl. Tighten hose clamp.

(7) Place the upper cowl in place and secure it to the nose cowl, lower cowlings and firewall flange with fasteners.

C. Cleaning, Inspection and Repair

(1) Clean the cowling with a suitable solvent and then wipe dry with a clean cloth.

(2) Inspect the cowling for dents, cracks, loose rivets, damaged or missing fasteners or screws, and damaged areas.

(3) Repair all defects to prevent further damage.

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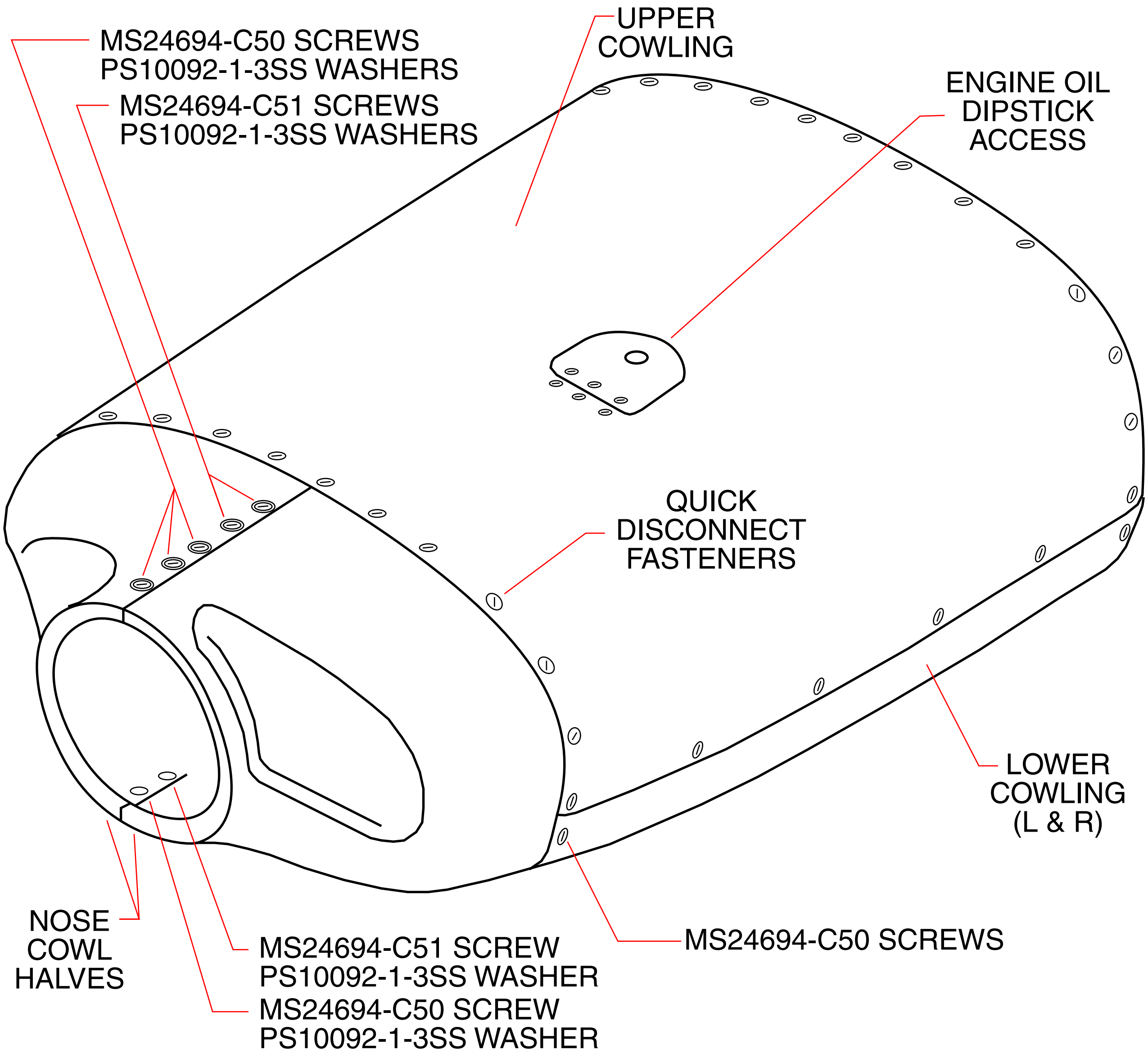

Engine Cowling
Figure 1

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MOUNTS

1. Engine Mount

   A. Inspection

      (1) 100 Hours

NOTE: This inspection incorporates the requirements of Piper Service Bulletin No. 1103F.

For any airplane with engine mount P/N 89137-041 or P/N 89137-042 installed and the engine mount has accumulated 200 hours time-in-service, each one hundred (100) hours time-in-service perform the following inspection.

NOTE: Installation of engine mount P/N 89137-043 relieves this repetitive inspection requirement.

(a) Clean the nose gear actuator attach feet area (see “Figure 1 (Sheet 1 of 2)”).

(b) Completely remove all of the paint or Dinitrol AV8 and AV30, if the paint was previously removed and covered with Dintrol AV8 and AV30, from the inspection area (see “Figure 1”.

NOTE: Paint must be removed using chemical processes only. The use of abrasives or other mechanical methods to remove the paint will hide the existence of any cracks, making an accurate inspection impossible. Use isopropyl alcohol to wipe clean the area of the engine mount where paint was removed.

(c) Perform a fluorescent penetrant inspection of the nose gear actuator attach feet for cracks per AC 43.13-1B, Chapter 5, Section 5. Inspect the surfaces identified in “Figure 1” with emphasis on welded areas.

NOTE: This fluorescent penetrant inspection shall be performed only by personnel with “Level I Special” qualification or higher, as described in Advisory Circular 65-31B, “Training, Qualification, and Certification of Nondestructive Inspection Personnel.”

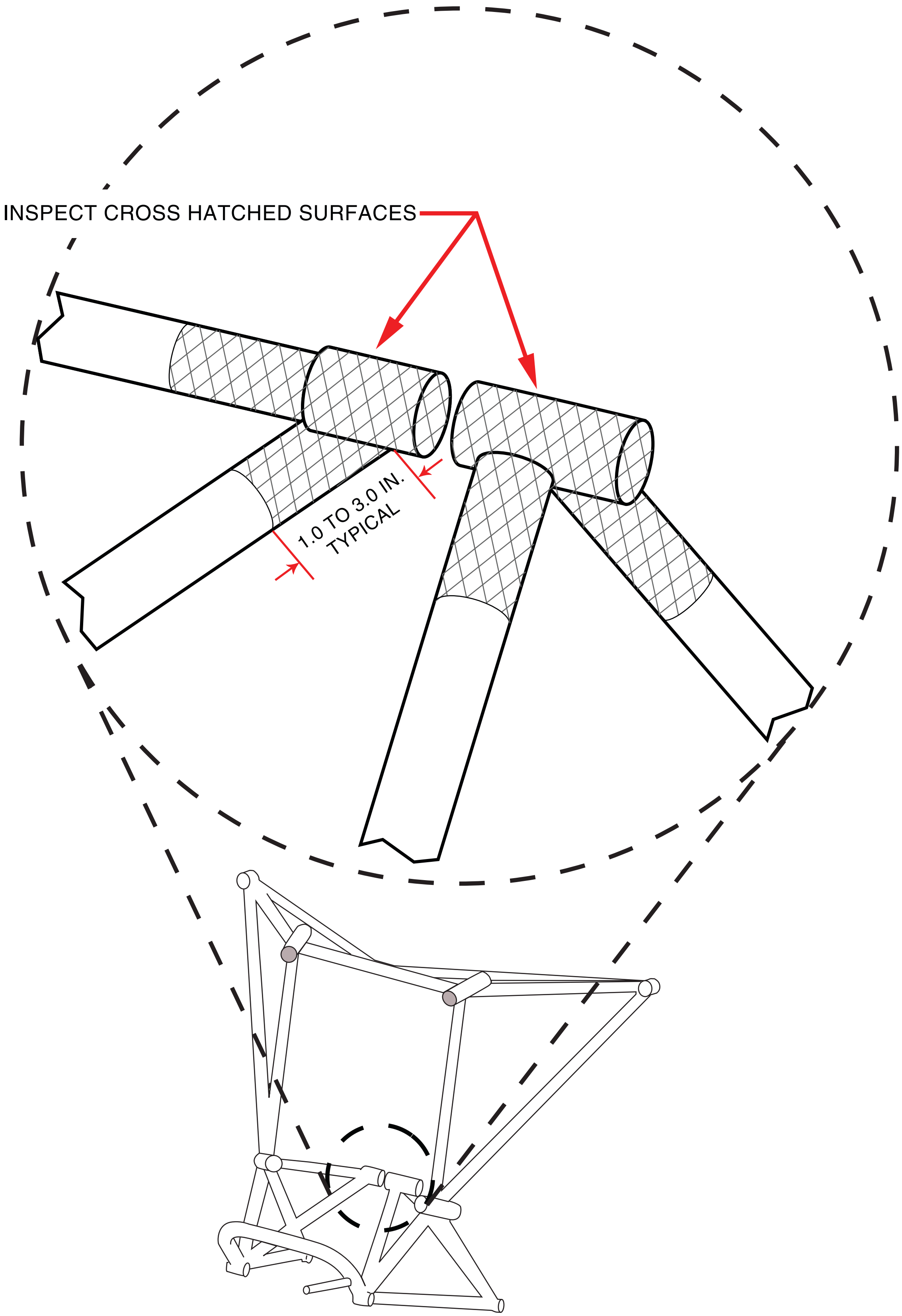
CAUTION: FLIGHT IS PROHIBITED WITH A CRACKED ENGINE MOUNT.

(d) If a crack is detected, the engine mount must be replaced prior to further flight. See “Removal” and “Installation” on page 71205, below.

(e) If no cracks are found, the engine mount can continue in service until the next 100 hour inspection.

(f) Clean the actuator attach feet and apply a two-coat corrosion prevention compound (CPC) to the area where the paint was removed. This two-coat CPC consists of Dinitrol/Ardrox AV8 as a primer coating and (after the AV8 has dried) Dinitrol/Ardrox AV30 as a top coating.

(g) Make an appropriate logbook entry documenting this inspection.



100 Hour Engine Mount Inspection
Figure 1 (Sheet 1 of 2)

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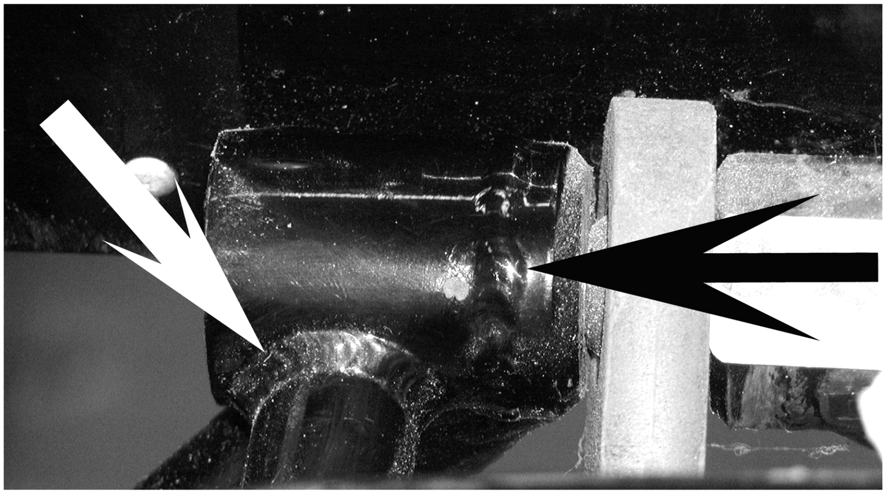
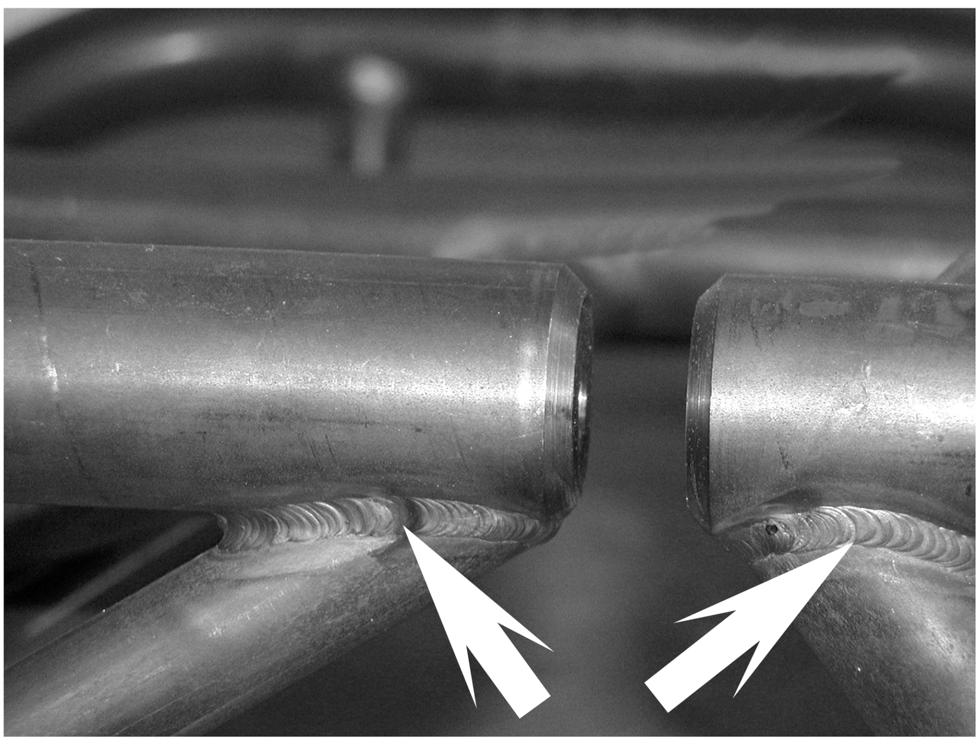


OLD STYLE ATTACH FEET
(89137-041 up to 2002)



NEW STYLE ATTACH FEET
(89137-041 2002 and later; and 89137-042)

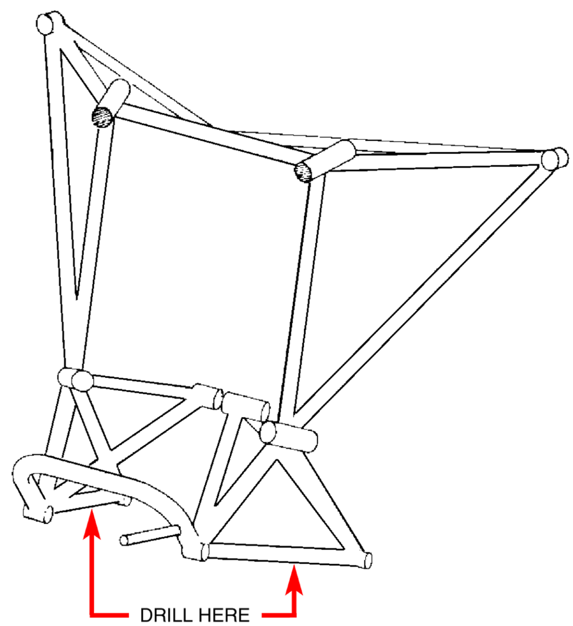
100 Hour Engine Mount Inspection
Figure 1 (Sheet 2 of 2)

(2) Corrosion Inspection, Immersion in Water

The following guidance is general in nature and should be applied or varied to fit the individual situation based on water level during immersion, length of time immersed, length of time since exposure, etc. Proceed as follows:

(a) Inspection

1) Level the aircraft per 8-20-00.
2) In the two lower tubes as indicated in “Figure 2”; drill a 3/16 inch hole in each tube, at the approximate mid-point.
3) Visually inspect the interior surface of each tube through the 3/16 inch hole for evidence of internal corrosion.
4) Should evidence of corrosion be detected in step (c), above, replace the engine mount. If no corrosion is detected, proceed with corrosion prevention, paragraph (2), below.



Engine Mount Corrosion Inspection Points
Figure 2

(b) Corrosion Prevention

If no evidence of corrosion is detected in step (3), above, proceed as follows:

1) Place a drip pan below the inspection holes in each engine mount tube.
2) Insert a plastic tube thru each inspection hole and feed it up to the high point of the engine mount tube.
3) Using a syringe inserted into the end of the plastic tube, pump linseed oil into the upper end of the engine mount tube while rotating the syringe / plastic tube assembly to assure maximum coverage. Continue pumping until the lower end of the engine mount tube is filled with linseed oil to the level of the inspection hole.
4) Now, draw the plastic tube out of the upper end of the engine mount tube and reinsert it in the opposite direction, feeding it to the lower end of the engine mount tube.
5) Suck excess linseed oil out of the engine mount tube with the syringe / plastic tube assembly.
6) When linseed oil can no longer be picked up by the syringe / plastic tube assembly, remove it and allow the engine mount tube to drain into drip pans for approximately two hours.
7) Purge excess oil from tubes by applying air pressure to each 3/16 inch inspection hole, one at a time.
8) Ensure that roughly the same amount of linseed oil that was pumped in is retrieved in the drip pans.
9) Apply a liberal coating of an approved fuel tank sealant (see Chart 3, 91-10-00) to each inspection hole and seal the hole with an appropriate blind rivet. After installing the rivet, apply a liberal coating of the approved fuel tank sealant over the head of the rivet.

B. Removal

(1) Remove engine per “Removal” on page 71007 under Engine in 71-00-00.

(2) Remove nose landing gear actuator per Removal under Nose Gear Actuator Assembly in 32-30-00.

(3) Remove nose landing gear per Removal under Nose Gear Assembly in 32-20-00.

(4) Support engine mount using an engine lift or other suitable hoist.

(5) Remove lower right engine mount bolt, washer, and nut, and remove the assist spring and spacers. Note positioning of spacers and assist spring rod end for reinstallation.

(6) Remove remaining three engine mount bolts, washers, and nuts and remove engine mount.

C. Installation

WARNING: IN S/N’S 4636001–4636620 & S/N’S 4692001–4692207, FIRST TIME INSTALLATION OF A SERVICE REPLACEMENT ENGINE MOUNT P/N 89137-043 REQUIRES MODIFICATION OF THE AIRPLANE’S WEIGHT AND BALANCE RECORD. ENGINE MOUNT P/N 89137-043 ADDS 1.25 LBS AT ARM 78.00.

CAUTION: WHEN INSTALLING A REPLACEMENT ENGINE MOUNT ASSEMBLY, ENSURE IT CONFORMS TO LATEST DESIGN AS SHOWN IN “Figure 1”, SHEET 2.

(1) Position engine mount in place using an engine lift or other suitable hoist.

(2) Install two upper and the lower left engine mount bolts, washers, and nuts and hand tighten.

(3) Install lower right engine mount bolt, washer, and nut, and the assist spring and spacers. Position spacers and assist spring rod end as noted during removal.

(4) Torque all four engine mount bolts 240–270 in.-lbs.

(5) Install nose landing gear per Installation under Nose Gear Assembly in 32-20-00.

(6) Install nose landing gear actuator per Installation under Nose Gear Actuator Assembly in 32-30-00.

(7) Install engine per “Installation” on page 71008 under Engine in 71-00-00.

2. Engine Shock Mounts Replacement

The engine shock mounts may be replaced with the engine installed as well as removed from the airplane. Refer to “Figure 1”, 71-00-00, for the arrangement of the shock mount assemblies. The top shocks are assembled so the silver colored shocks are aft and the gold colored shocks are forward. The lower shock mounts are installed opposite of the top shock mounts. The procedure described in this paragraph is with the engine installed.

A. Remove the engine cowling.

B. Attach a one-half ton (minimum) hoist to the engine hoisting hooks and relieve tension from the shock mounts.

C. Loosen the upper shock mount attachment nuts.

D. Remove the lower mount attachment nuts, washers, forward shock mounts and spacers.

E. Remove the lower attachment bolts just far enough to allow the aft shock mounts to be removed. The bushing in each lower mount must be removed with the bolt.

CAUTION: CARE SHOULD BE TAKEN NOT TO INTRODUCE ADVERSE STRESSES ON THE CONTROL CABLES, ELECTRICAL CABLES, HOSES AND OTHER ITEMS ATTACHED TO THE ENGINE, WHILE HOISTING THE ENGINE.

F. Raise the nose of the engine enough to remove the lower aft shock mounts and replace with new ones.

G. Lower the engine, slide the attachment bolts with bushings into place and install the spacers, forward shock mounts, washers and nuts. Start nuts only a few threads.

H. Rotate the heat shields on the lower mounts to provide the greatest protection against exhaust heat.

I. Remove the upper mount attachment bolts, nuts, washers, forward shock mounts and spacers.

J. Lower the engine enough to replace the upper aft shock mounts. Raise the engine into position.

K. Install the spacers, forward shock mounts, mounting bolts, washers and nuts.

L. Tighten engine shock mounts (4) to torque specified in “Figure 1”, 71-00-00.

DRAINS

Cylinder Head Drain

The cylinder head drain valve is located on the bottom forward right hand air intake plenum. The drain valve is connected to the cylinder head drain lines, which come from both banks of cylinders. A drain tube goes from the drain valve through the lower cowling.

The drain valve is closed by manifold pressure when the engine is running. When the engine is shut down, the lack of manifold pressure allows the drain valve to open, thus draining any excess fuel from the cylinder heads.

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CHAPTER

73

ENGINE FUEL & CONTROL

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CHAPTER 73

LIST OF EFFECTIVE PAGES

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CHAPTER 73 - ENGINE FUEL AND CONTROL

TABLE OF CONTENTS

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DISTRIBUTION

1. Fuel Injection System

A. Description

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

These airplanes are powered by a Lycoming (TIO-540-AE2A) engine that is fuel injected to achieve even fuel distribution to all cylinders, for a smooth and efficient running engine.

A vaned, positive displacement, engine-driven fuel pump supplies fuel under pressure to the fuel injector servo metering unit. This metering unit determines the proper amount of fuel to be sent to the flow divider where the fuel flow is distributed equally among the six injector nozzles.

To atomize the fuel in the injector nozzles prior to entering the cylinder, turbocharged compressed air pressure (deck pressure) is picked up just prior to the throttle valve injector metering unit and distributed evenly to each injector nozzle, where it atomizes the fuel prior to entering the cylinders. The nozzles continuously spray and atomize the metered fuel into the intake valve port of the engine cylinder heads in proportion to induction airflow.

B. Maintenance

(1) Little attention is required between scheduled injector overhauls. Check the following items during periodic inspection of the engine:

(a) Check tightness and lock of all nuts and screws which fasten the injector to the engine.

(b) Check all fuel lines for tightness and evidence of leakage. A slight fuel stain adjacent to the air bleed nozzles is not cause for concern.

NOTE: See latest revision Lycoming Service Bulletin 342, Fuel Lines and Support Clamp Inspection and Installation.

(c) Check throttle and mixture control rods and levers for tightness and lock.

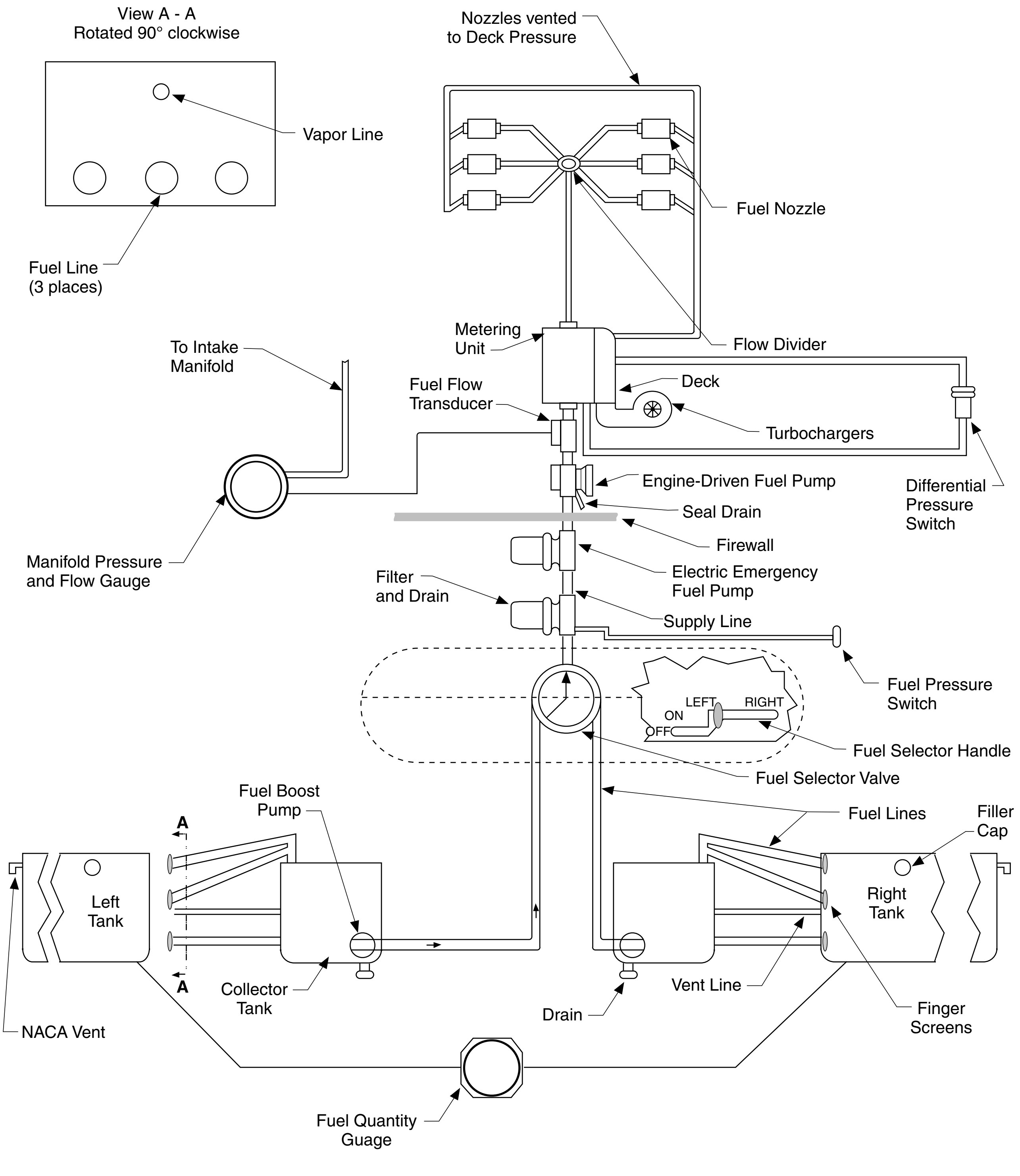
(d) Remove and clean the injector fuel inlet strainer at the first 25 hour inspection and each 50 hour inspection thereafter. Damaged strainer O-rings should be replaced.

(2) Tests prove that gasoline, which becomes stale due to prolonged storage, absorbs oxygen rapidly. This stale oxidized gasoline acquires a very distinctive odor similar to varnish, causes rapid deterioration of synthetic rubber parts, and forms a gummy deposit on internal metal parts. This condition does not occur during normal operation of the injector where fresh fuel is being constantly circulated.

C. Lubrication

(1) The clevis pins used in connection with the throttle and manual mixture control levers should be checked for freedom of movement and lubricated, if necessary. There is no need for lubrication of the injector in the field between scheduled overhauls.

(2) Place a drop of engine-grade oil on the end of the throttle shaft in such a manner that is can work into the throttle shaft bushings.



Fuel System Fluid Diagram
Figure 1

2. Fuel Injector

A. Removal

(1) Remove the lower cowl panel.

(2) Disconnect the throttle and mixture control cables at the injector. Disconnect the control rod that comes from the absolute pressure controller.

(3) Disconnect the fuel inlet, flow, pressure, and discharge lines at the injector.

(4) Remove the bolts securing the injector to the air box and induction duct.

(5) Remove the fuel injector.

B. Installation

(1) Attach the injector to the air box and induction duct.

(2) Connect the fuel inlet, flow, pressure, and discharger lines to the injector.

(3) Connect the throttle and mixture control cables to the injector. Attach the control rod that comes from the absolute pressure controller to the fuel injector. Rig controls.

(4) Pressure-check for leaks.

(5) Adjust idle speed and mixture.

(6) Replace cowling.

C. Storage Preparation

CAUTION: BECAUSE FLUID CAN EASILY ENTER THE AIR SECTION OF THE INJECTOR THROUGH THE IMPACT TUBES OR THE ANNULAR GROOVE AROUND THE VENTURI, INSTALL A PROTECTIVE PLATE ON THE SCOOP MOUNTING FLANGE WHEN PERFORMING ROUTINE MAINTENANCE ON THE ENGINE, SUCH AS WASHING DOWN THE ENGINE AND AIR SCOOP, SERVICING THE AIR FILTER (SURPLUS OIL ON THE ELEMENT), OR WHEN INJECTING PRESERVATIVE INTO THE ENGINE PRIOR TO STORING OR SHIPPING.

Any unit taken out of service, or units being returned for overhaul, must be flushed with preserving oil (Specification MIL-0-6081, Grade 1010), using the following procedure:

CAUTION: DO NOT EXCEED 15 PSI AIR PRESSURE AS INTERNAL DAMAGE TO THE INJECTOR MAY RESULT.

(1) Remove plugs and drain all fuel from the injector. If available, apply 10 - 15 psi air pressure to the fuel inlet until all fuel is discharged from the injector.

(2) Replace plugs and apply flushing oil filtered through a 10 micron filter at 13 - 15 psi to the injector fuel inlet until oil is discharged from the outlet.

(3) Replace fuel inlet shipping plug.

(4) After filling with preservative oil, the injector should be protected from dust and dirt and given such protection against moisture as climactic conditions at the point of storage require. In most cases, storing the unit in a dry area will be sufficient.

(5) If the unit is to be stored near or shipped over salt water, the following precautions should be observed:

(a) Spray the exterior of the injector with an approved preservative oil.

(b) Pack in a dustproof container. Wrap the container with moisture and vapor-proof material and seal. Pack the wrapped unit in a suitable shipping case. Pack a one-half pound bag of silica gel crystals in the dustproof container with injector. The bag must not touch the injector.

3. Fuel Air Bleed Nozzle

A. Removal

The nozzles must be carefully removed as they or the cylinders may be damaged.

(1) Remove the lower engine cowl.

(2) Disconnect the fuel line from the nozzle.

(3) Remove the spring retainer and spring from the nozzle stem.

(4) Disconnect the nozzle shroud from the vent hose and remove it from the nozzle.

(5) Carefully remove the nozzle, using the correct size deep socket.

B. Installation

(1) Install the nozzles and torque to 60 inch - pounds.

(2) Ascertain that the O-rings are properly installed on the nozzle stem and install the nozzle shroud (refer to “Figure 2”).

(3) Connect the vent to the nozzle shroud.

(4) Install the spring and spring retainer on the nozzle stem.

(5) Connect the fuel line to the nozzle and clamp the fuel lines as described in latest revision of Lycoming Service Bulletin No. 335.

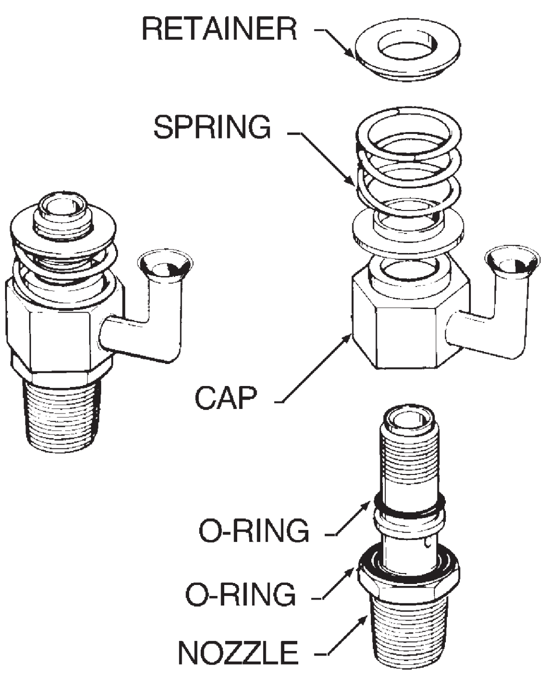
(6) Install the engine cowl.

C. Cleaning and Inspection

(1) Clean the nozzle with acetone or equivalent and blow out all foreign particles. Do not use wire or other hard objects to clean orifices. (Refer to the latest revision of Lycoming Service Instruction No. 1275)

(2) Inspect and replace nozzle O-rings if found to be cracked, brittle or distorted.

(3) A test procedure for air bleed nozzles is described in latest revision of Lycoming Service Instruction No. 1275.



Fuel Air Bleed Nozzle
Figure 2

INDICATING

NOTE: PA-46-350P S/N,s 4636021–4636374 incorporate an Integrated Engine Instrumentation System. See Section 77-40-00 for more information.

NOTE: In PA-46-350P S/N's 4636375 and up; and PA-46R-350T S/N's 4692001 and up; engine data is collected by the Data Acquisition Unit (DAU) (if Avidyne-equipped) or Engine/Airframe Unit (GEA) (if Garmin G1000/G1000 NXi-equipped), and is normally displayed on the Multi-Function Display (MFD). In the event of a MFD failure, all engine instruments can be displayed on the Primary Flight Displays (PFDs). See 34-22-00 (Avidyne), 34-25-01 (G1000), or 34-25-02 (G1000 NXi); 77-40-00, and 91-77-40 for more information.

1. Fuel Flow Gauge

The fuel flow gauge is mounted to the right of center in the upper section of the instrument panel, and is electrically operated by the flow transducer.

A. Troubleshooting

See "Chart 1".

B. Removal

(1) Gain access to the instrument from behind the instrument panel. Disconnect the electrical connection from the instrument.

(2) Remove the post light(s) by turning off nut.

(3) Remove the screws securing the instrument to the panel.

(4) Remove the instrument from the panel.

C. Installation

(1) Place the instrument in its proper panel cutout and secure with screws.

(2) Install the post light(s) and secure. Do not overtighten the nut.

(3) Connect the electrical connector to the gauge.

2. Fuel Pressure Low Warning Light

Illumination of the FUEL PRESS annunciator light indicates impending fuel starvation, due to insufficient fuel pressure to the engine. See also 31-50-00 and 91-31-50.

3. Emergency Fuel Pump

A. Adjustment

To adjust the emergency fuel pump output pressure:

(1) Loosen the jamnut on the adjustment screw to allow free movement.

(2) Adjust the pump pressure by turning screw in or out.

(3) Tighten fuel pump jamnut and perform pressure operational check.

CHART 1
TROUBLESHOOTING FUEL FLOW GAUGE

Trouble	Cause	Remedy
Pointer oscillates.	Air in fuel line.	Purge line.

B. Pressure Operational Check

WARNING: MAKE SURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR OF THE PROPELLER ROTATIONAL AREA DURING ENGINE START AND RUN-UP.

CAUTION: SHOULD EXCESSIVE FUEL PRESSURE OCCUR, IMMEDIATELY SHUT DOWN ENGINE AND ADJUST FUEL PUMP.

(1) Pull the Left or Right boost pump circuit breaker, depending upon fuel tank selection.

(2) Run engine to maximum power and observe fuel pressure gauge for 38 ± 2 psi.

(3) Reset the appropriate boost pump circuit breaker.

(4) Shut down engine.

4. Fuel Flow Transducer

The fuel flow transducer is located on the engine accessory case just below the left magneto.

A. Removal

(1) Disconnect electrical leads going to transducer.

(2) Disconnect B-nut on transducer fuel outlet line.

(3) Remove the nuts, washers, and bolts securing transducer to mounting bracket.

(4) Remove transducer mounting bracket from engine.

(5) Disconnect transducer from fuel pump.

B. Installation

(1) Coat all pipe threads with PST 563 sealant. Do not coat the orifice in the parts.

(2) Position transducer in place. Connect fuel IN port to fuel pump.

(3) Install transducer mounting bracket to engine.

(4) Attach transducer to mounting bracket with two each AN4-12A bolts, AN960-416 washers, and MS21042-4 nuts.

(5) Install AN816-6 adapter into transducer OUT port and connect B nut.

(6) Connect electrical leads to transducer.

CHAPTER

74

IGNITION

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CHAPTER 74

LIST OF EFFECTIVE PAGES

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CHAPTER 74 - IGNITION

TABLE OF CONTENTS

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ELECTRICAL POWER SUPPLY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This section provides only basic timing magnetos to the engine procedures and, if not provided in the appropriate vendor engine maintenance manual, magneto removal and installation instructions. Refer to the appropriate vendor's engine and magneto maintenance manuals for all other magneto maintenance procedures. See Introduction - Supplementary Publications.

1. Ignition System

A. Description

Ignition of the fuel charge in each cylinder is accomplished by two spark plugs independently excited by one of two Slick magnetos. Each magneto separately generates, times and distributes high tension (voltage) through leads to each cylinder. Both magnetos are pressurized by turbo compressor bleed air taken from the induction duct downstream from the right intercooler through an in-line filter to improve magneto efficiency at altitude.

The magnetos are controlled by two switches on the pilot's instrument panel. With the switch OFF, the magneto is grounded and will not produce spark. The right magneto fires the lower left and upper right spark plugs. The left magneto fires the lower right and upper left spark plugs.

The right magneto is standard and the left magneto is a retard breaker type with a fixed retard and long duration boosted spark for starting. A DC power source and a remote starting vibrator are required for operation. The magnetos incorporate an integral feed-through capacitor. The retarded magneto must have the right or standard magneto grounded during the starting cycle.

B. Troubleshooting

See "Chart 1" on page 74102.

C. Replacement Magnetos

NOTE: Check the magneto dataplate to verify the specific model number and series of the magneto being worked on.

D. Overhaul

Overhaul is required as conditions indicate, but in no case may a Slick 6300 series magneto's time-in-service exceed the TBO for the engine. Magnetos must also be overhauled after a lightning strike or following a sudden engine stoppage.

NOTE: An alternative to overhaul is complete magneto replacement with a new Slick magneto. New Slick magnetos incorporate all the latest design features and may be a cost-effective alternative to overhaul.

Information provided in this section is intended to support magneto removal, inspection, replacement and timing. For magneto overhaul procedures, see Slick's F-1100 Master Service Manual (see Introduction, Supplementary Publications, Vendor Publications, Magnetos).

CHART 1
TROUBLESHOOTING IGNITION SYSTEM

Trouble	Cause	Remedy
Failure of engine to start.	Defective spark plugs.	Clean and adjust or replace spark plugs.
	Defective ignition wire.	Check with electric tester and replace any defective wires.
	Improper operation of magneto.	Check timing of magnetos.
	Improper switch wiring for left magneto starting.	Reverse magneto switch wires.
	Frozen spark plug electrodes.	Replace spark plugs or dry out removed plugs.
	Shorted ignition switch or loose ground.	Check and replace or repair.
	Defective vibrator.	Replace vibrator.
Failure of engine to idle properly.	Faulty ignition system.	Check entire ignition system.
	Improper spark plug setting for altitude.	Check spark plug gap.
Low power and uneven running engine.	Defective spark plugs.	Clean or replace sparkplug.
	Magneto not working properly.	Check timing of magneto.
	Defective ignition wire.	Check wire with electric tester. Replace defective wire.
	Defective spark plug terminal connections.	Replace connectors or spark plug wire.
	Improper ignition timing.	Check magnetos for timing and synchronization.
Failure of engine to develop full power.	Faulty ignition.	Tighten all connections. Check system. Check ignition timing.
Rough running engine.	Lead deposit on spark plug.	Clean or replace plugs.

2. Magnetos

See “Figure 2” on page 74105.

A. 100 Hour Inspection

Every 100 hours or at annual inspection, whichever comes first, perform the following checks.

WARNING: BE SURE IGNITION SWITCH IS IN THE “OFF” POSITION AND THE CONDENSER P-LEAD IS GROUNDED.

(1) At annual or 100 hour inspections, check magneto to engine timing. If engine timing has advanced or retarded (timing drift) more than 4 degrees from the previous inspection set point or more than 5 degrees since original installation, refer to the Slick by Champion Master Service Manual, L-1363, for guidance on troubleshooting and correction.

If the timing drift is 4 degrees or less, re-time the magneto to the engine per “Adjust timing to engine” on page 74103. Record the set timing in the engine logbook inspection entry.

Note that timing drfit of more than 4 degrees within any 100 hour period warrants immediate investigation and correction.

EXAMPLE: On a 20 degree base timing engine, if the timing is found to be 15 degrees Before Top Dead Center (BTDC) or 25 degrees BTDC, in a 100 hour interval, this is considered excessive timing and should be investigated for cause.

NOTE: Champion employs a magneto design where the wear of breaker cam surfaces and point surface erosion offset each other, resulting in a larger drift in engine timing.

Timing retards when the cam surface wears more quickly than the point surfaces. Timing advances when the point surfaces erode more quickly than the cam.

When magneto timing drift exceeds 5 degrees, the magneto output may be diminished and ignition of the fuel-air mixture becomes less effective. Symptoms of this condition can include: hard starting, a rough running engine, and RPM/differental RPM drops exceeding the engine manufacturers allowable preflight mag checks.

(2) Adjust timing to engine

See “Figure 1” on page 74104.

(a) Turn the engine crankshaft in the normal direction of rotation until the No. 1 cylinder is in the full-advance firing position.

1) Cover spark plug hole of number one cylinder with thumb. Rotate crankshaft until pressure is felt on thumb.

2) Rotate crankshaft slowly until the advance timing mark on the starter ring gear is in alignment with the small hole located at the two (2) o’clock position on the front face of the starter housing. When the 20° mark on the gear is aligned with the small hole, number one piston is at 20° BTC.

NOTE: Verify correct engine timing for the airplane being worked on by checking the engine dataplate.

(b) Scribe a reference mark on the magneto mounting flange and engine accessory case.

(c) Loosen the magneto mounting bolts, and connect a standard timing light between engine ground and the magneto condenser terminal.

WARNING: DO NOT ROTATE PROPELLER WHEN IGNITION SWITCH IS IN THE "ON" POSITION. THE MAGNETOS WILL FIRE THE SPARK PLUGS IF THE PROPELLER IS ROTATED - FATAL INJURY IS POSSIBLE.

(d) Turn ignition switch ON.

(e) Rotate the magneto, in its mounting, in the direction of normal operating rotation until the timing light indicates the contact breaker points are open.

(f) Slowly rotate the magneto opposite normal rotation of the magneto on the engine mounting until the timing light (or audible signal) goes out.

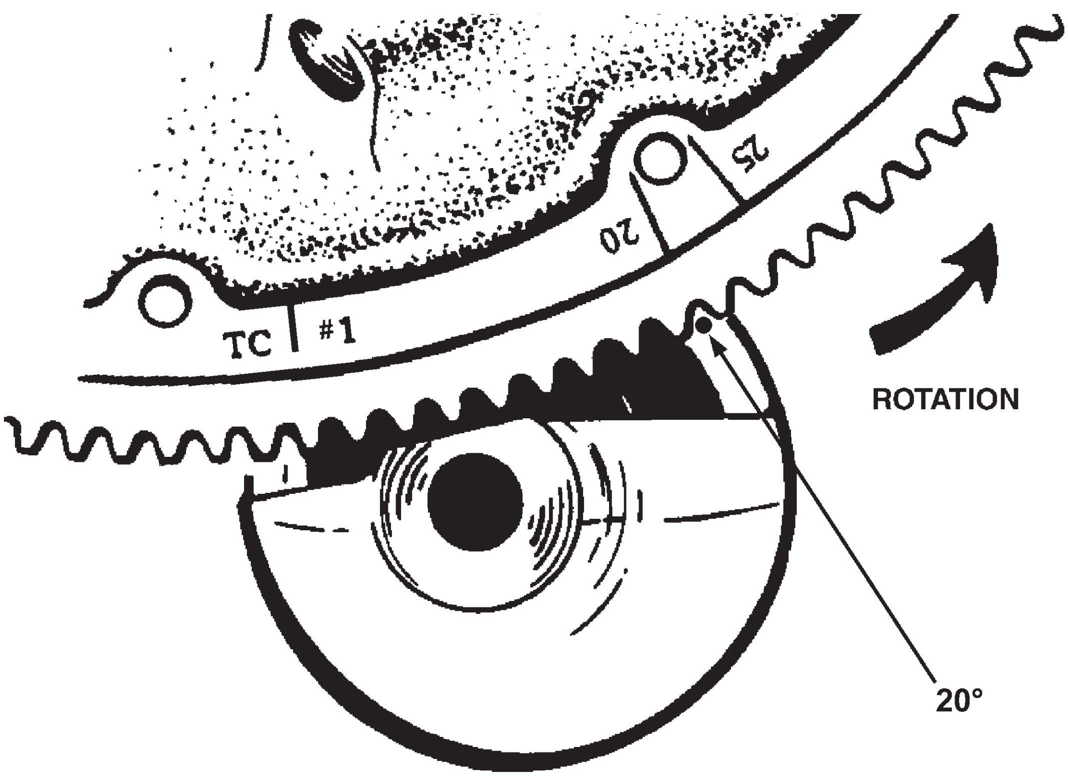
(g) Measure the distance from the reference mark previously scribed on the accessory case and the corresponding reference mark on the magneto. If this measurement is more than 1/8 inch, remove the magneto per "Removal" on page 74206, and inspect/adjust the contact breaker points per the Slick F-1100 Master Service Manual. A 1/8 inch change corresponds to an approximate 5° change in internal magneto timing.

(h) Secure the magneto in this position, alternately tightening the magneto mounting clamps - first to 8 ft-lbs. and finally to 17 ft-lbs. of torque.

(i) Turn ignition switch OFF.

(3) Inspect harness. See Section 74-20-00.

(4) Inspect P-lead attachment. The P-lead connects the magneto primary circuit to the ignition switch. If the P-lead is disconnected, the magneto will be "HOT" and will fire the spark plug if the propeller is rotated. Verify that the P-lead is attached to the condenser stud. Torque to 13–15 in·lbs.

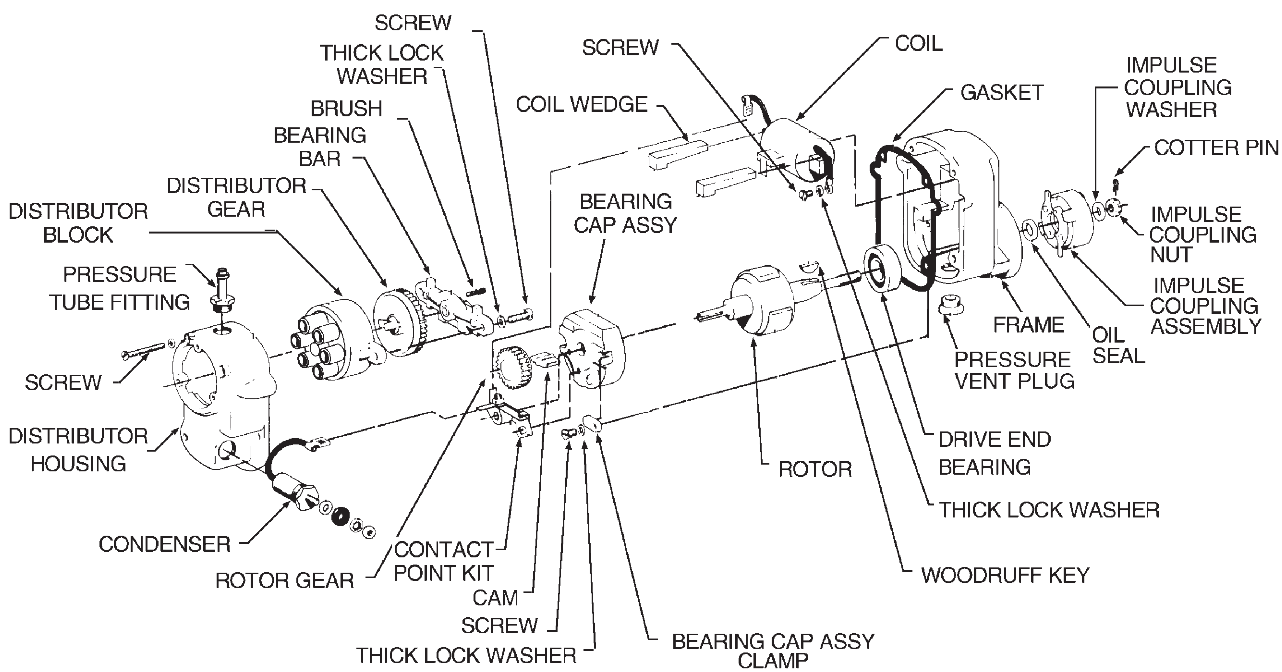


Engine Timing Marks (Typical)
Figure 1

(5) Inspect switch wire (left magneto only). This retard breaker lead connects the retard contact points to the ignition vibrator. If disconnected, the starting circuit will be inoperative. Torque to 13–15 in·lbs.

(6) Inspect and clean inlet nozzle. Yellow or white particles or any oily film indicates moisture contamination and possible lack of pressurization. Inspect and repair pressurization system.

(7) Inspect and clean orifice vent. Maximum orifice diameter is .025 inch.



Exploded View of Magneto
Figure 2

B. Removal

CAUTION: ASCERTAIN THAT THE PRIMARY CIRCUIT OF THE ENGINE IS GROUNDED BEFORE WORKING ON THE ENGINE.

Before removing the magnetos, make sure the magneto switches are OFF.

WARNING: THE MAGNETO IS NOT INTERNALLY GROUNDED, WHEN THE GROUND LEAD IS DISCONNECTED THE MAGNETO IS HOT. REMOVING THE HARNESS ASSEMBLY FIRST AND INSTALLING THEM LAST, MINIMIZES THE DANGER OF STARTING THE ENGINE ACCIDENTALLY WHEN THE GROUND LEAD IS REMOVED FROM THE MAGNETO.

(1) Turn the engine crankshaft in the normal direction of rotation until the No. 1 cylinder is in the full-advance firing position.
(2) Remove the harness cap from the magneto. Before doing this, place an index mark on the harness cap and distributor housing to ensure proper alignment upon reassembly.
(3) Disconnect the P-lead and pressurization tube from magneto. Disconnect the retard breaker lead to the starting circuit from the left magneto.
(4) Remove the nuts, washers and clamps, and remove the magnetos from the engine.
(5) Cover the magneto accessory opening with suitable material to prevent internal engine contamination.

C. Installation

WARNING: BE SURE SWITCH IS IN OFF POSITION AND THE P LEAD IS GROUNDED.

When installing new or adjusting breaker points and before timing the magneto to the engine, it is important that the internal timing of the magneto be correct. To find number one tower, the following instructions should be performed:

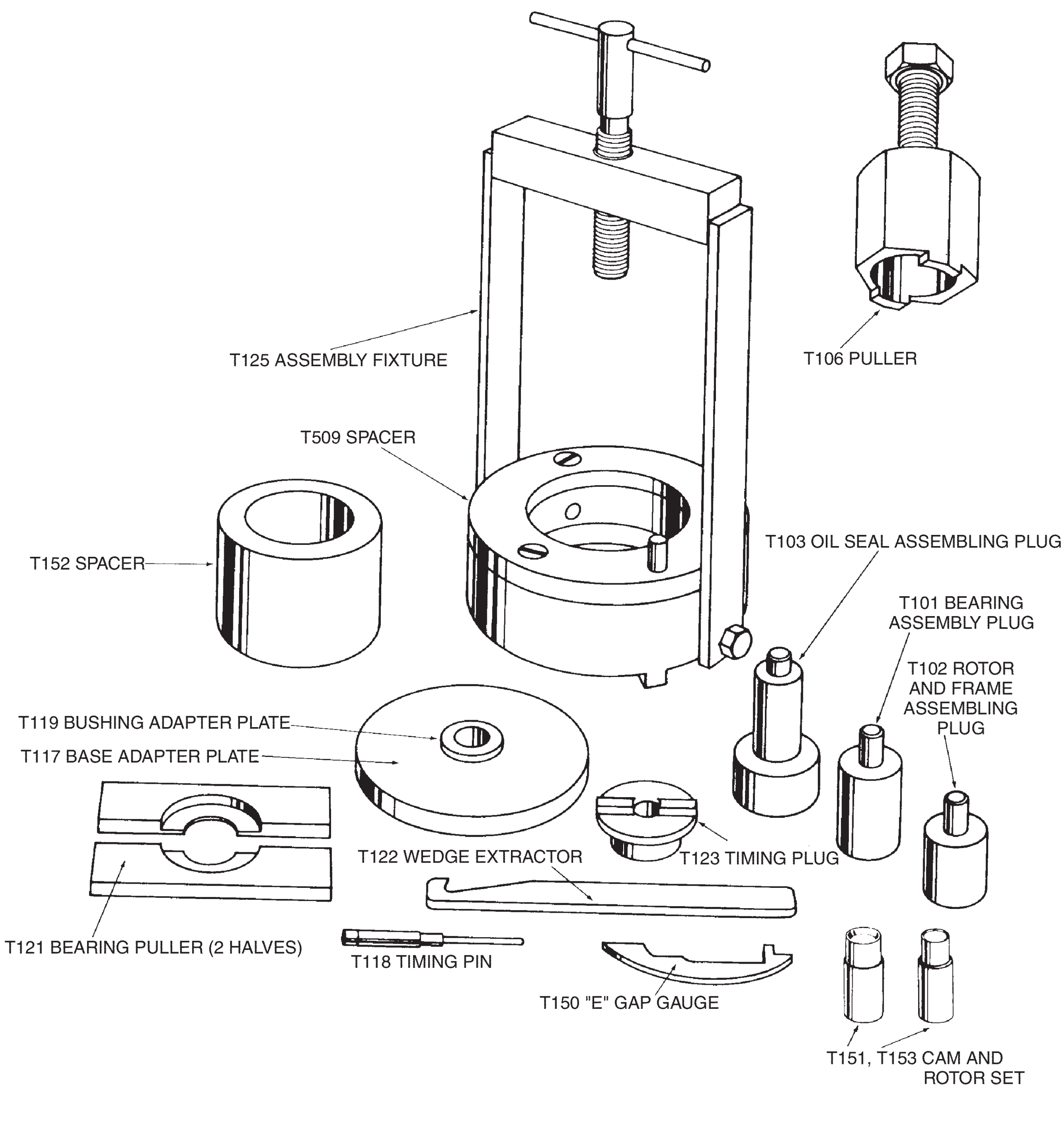
NOTE: No need to spark out these magnetos.

(1) Insert the T-118 timing pin (“Figure 3”) in the L or R hole in the distributor block (depending on rotation of the magneto).
(2) Turn rotor opposite the rotation of the magneto until the pin engages the gear.
(3) If the pin is binding and will not go in the hole in the gear, you have hit the pointer on the gear. Pull the pin out, enough to continue opposite rotation until the pointer has passed, re-insert pin.
(4) When the pin sticks through the hole in the gear about 1/4 inch, you are now ready to fire number one cylinder.
(5) Turn the engine crankshaft in the normal direction of rotation until the No. 1 cylinder is in the full-advance firing position.
    (a) Cover spark plug hole of number one cylinder with thumb. Rotate crankshaft until pressure is felt on thumb.
    (b) Remove plug in front of number six cylinder. Rotate crankshaft slowly to observe timing mark on alternator drive gear. When the mark on the gear is centered in the viewing hole, number one piston is at 20° BTC.

    NOTE: Verify correct engine timing for the airplane being worked on by checking the engine dataplate.

(6) Place a new gasket on magneto flange. Install magneto carefully so drive coupling lugs mate with slots of drive bushings. Install holding washers, lockwashers, and nuts.

    NOTE: Do not tighten completely. Allow for turning magneto for final timing.



Slick T-100 Assembly and Timing Tool Kit
Figure 3

(7) After the magneto is installed on engine, remove the timing pin. The magneto is now ready to be timed to the engine.

(8) Complete magneto to engine timing procedure listed under 100 Hour Inspection, above.

WARNING: THE MAGNETO IS NOT INTERNALLY GROUNDED, WHEN THE GROUND LEAD IS DISCONNECTED THE MAGNETO IS HOT. REMOVING THE HARNESS ASSEMBLY FIRST AND INSTALLING IT LAST, MINIMIZES THE DANGER OF STARTING THE ENGINE ACCIDENTALLY WHEN THE GROUND LEAD IS REMOVED FROM THE MAGNETO.

(9) Replace the harness cap onto the magneto. Align the index marks made on the harness cap and distributor housing when removed.

(10) Connect the P-lead and pressurization tube to magneto. Connect the retard breaker lead to the starting circuit to the left magneto.

DISTRIBUTION

1. Harness

A. Inspection

(1) Check lead assemblies for nicks, cuts, mutilated braiding, badly worn section or any other evidence of physical damage. Inspect spark plug sleeves for chafing or tears and damage orstripped threads on coupling nuts. Check compression spring to see if it is broken or distorted. Inspect grommet for tears. Check all mounting brackets and clamps to see that they are secure and not cracked.

(2) Using an ohmmeter, buzzer, or other suitable low voltage device, check each lead for continuity. If continuity does not exist, wire is broken and must be replaced.

(3) For electrical test of harness assembly, use a high voltage, direct current tester such as the TAKK Model 86 or 86A or an equivalent direct current high voltage tester capable of delivering a test potential of 10,000 volts. Connect ground lead to high voltage tester to outer shielding braid of a single lead. Connect plug terminal. Turn tester ON and apply 10,000 volts. The insulation resistance should be 100 megohms minimum. Proceed to check other leads of harness in same manner.

(4) Minor repair of the harness assembly, such as replacement of contact springs, spring retainer assemblies, insulating sleeves or of one lead assembly, can be accomplished with the harness assembly mounted on the engine. However, should repair require replacement of more than one lead assembly or of a cable outlet plate, the harness should be removed from the engine and sent to an overhaul shop.

B. Removal

(1) Disconnect the clamps that secures the wires to the engine and accessories.

(2) Loosen the coupling nuts at the spark plugs and remove the insulators from the spark plug barrel well. Use caution when withdrawing the insulator not to damage the insulator spring.

(3) Place a guard over the harness insulators.

(4) Remove the harness assembly terminal plate from the magneto.

(5) Remove the harness from the airplane.

C. Installation

Before installing harness on magneto, check mating surfaces for cleanliness.

(1) Place the harness terminal plate on the magneto and tighten nuts around the plate alternately to seat cover squarely on magneto.

NOTE: The right magneto fires the lower left and upper right spark plugs. The left magneto fires the lower right and upper left spark plugs.

(2) Route ignition wires to their respective cylinders per markings on wire insulation.

(3) Clamp the harness assembly in position and install the magneto terminal plate.

(4) Connect the leads to the spark plugs.

D. Disassembly

(1) To remove spring from a damaged lead, turn the spring counterclockwise while pulling gently. This will remove the spring and the M-1498 Electrode Screw from the end of the coiled conductor.

(2) To separate the spring and the screw, hold the electrode screw with a pair of pliers and turn the spring clockwise until it is through the threaded portion.

(3) Remove the insulator sleeve from the end of the wire.

(4) To remove a lead from the M-1568 Harness Cap, use diagonals or cutting pliers and cut the lead off close to the housing. A drift or punch can be used to tap the ferrule loose from the housing.

NOTE: Further service on the Slick harness wires requires the use of Slick M-1495 Service Tool Kit, obtained from:

Slick Aircraft Products
(See Introduction, Supplementary Publications, Vendor Publications, Magnetos.)

E. Assembly

(1) Cut a piece of harness wire to the length required. Do not stretch the wire when measuring it.

(2) On the magneto end, make a final mark 0.75 inch from the end of the wire. A 0.937 inch mark should be made from the spark plug end of the wire.

(3) Flare out shielding, then without allowing any of the shielding to fold under, insert Slick M- 1743 Stripping Tool under the braided shielding (refer to “Figure 1”).

(4) Make sure the stripping tool is inserted past the cutting mark, then cut the shielding with a sharp knife using a rolling motion and remove the shielding and stripping tool. Take care not to cut the silicone insulation (refer to “Figure 2”).

(5) Cut exposed insulation 0.062 inch back from end and roll the insulation clockwise to remove it. Do not pull the insulation off the wire. Trim the end of the coiled conductor to make a clear hole for inserting the stud (refer to “Figure 3”).

(6) Using M-1742 Pin Vise, insert M-1741 Drill (72 drill), drill out the silicone rubber from inside of coiled conductor approximately 0.375 inch deep (refer to “Figure 4”).

(7) On spark plug end of wire install M-1462 Nut followed by M-1459 Female Taper Hex Ferrule (refer to “Figure 5”).

(8) After installation of nut and ferrule, bend and rotate the silicone insulation as illustrated in Figure 6 to flare out the shielding so a drive ferrule can be inserted. Take care not to cut the silicone insulation with the sharp braiding while the wire is being rotated (refer to “Figure 6”).

(9) On spark plug end of wire install M- 1458 Male Tapered Drive Ferrule over silicone insulation and under shielding to within 0.062 inch from flange of ferrule. Make certain that shielding is away from ferrule flange then slide Ferrule M-1459 over the M-1458 Drive Ferrule until tight (refer to “Figure 7”).

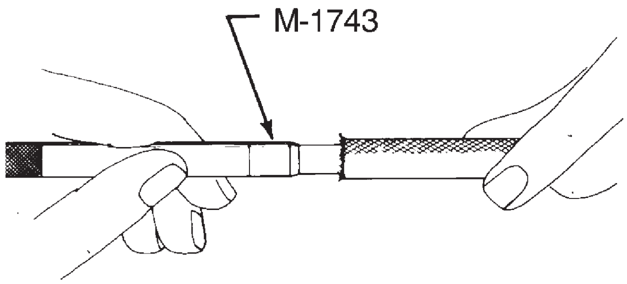
NOTE: Do not reuse the M-1458 Drive Ferrule.

(10) For spark plug end, mount M-1747 Drive Plate in a bench vise. Set the hex ferrule in the slot of the drive plate. Drive the M-1458 Drive Ferrule flush against the hex ferrule using the M-1744 Drive Tool (refer to “Figure 8”).

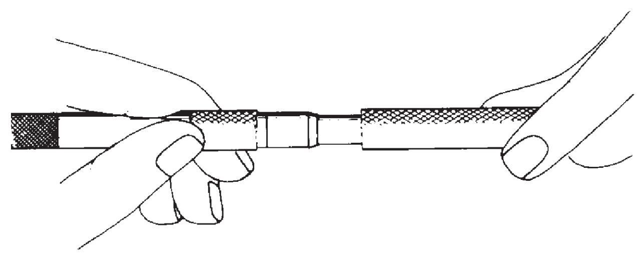
(11) For magneto end of wire, insert wire through approximate hole in the M-1568 Harness Cap so the shielding is through the hole as shown in “Figure 9”.

(12) Install an M-1458 Male Tapered Drive Ferrule over insulation and under shielding as in Step (9), then drive the ferrule into the M-1568 Harness Cap using the M-1744 Drive Tool, similar to Step (10) (refer to “Figure 10”).

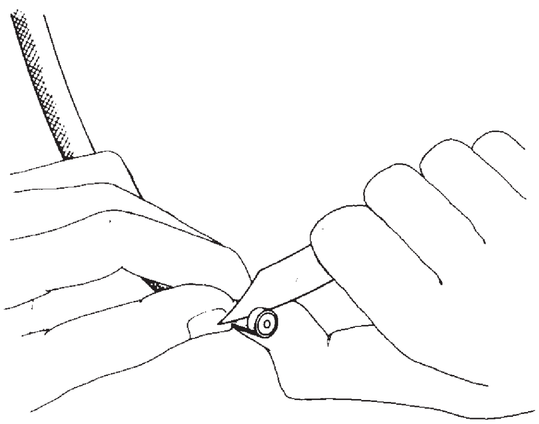
(13) Clamp the threaded end of the M-1498 Electrode Screw in the M-1742 Pin Vise. Insert the tapered pin of the electrode screw into the center of the coiled conductor by turning the pin vise counterclockwise and pushing at the same time until the screw is flush with the insulation. This is done at both ends of the wire assembly (refer to “Figure 11”).



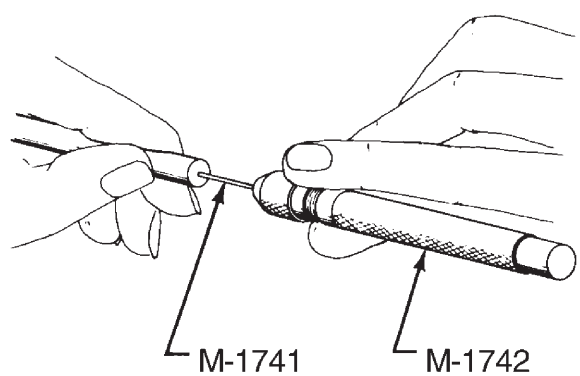
Stripping Tool
Figure 1



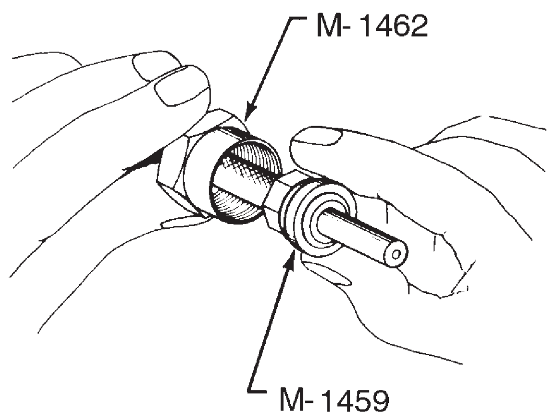
Inserting Stripping Tool
Figure 2



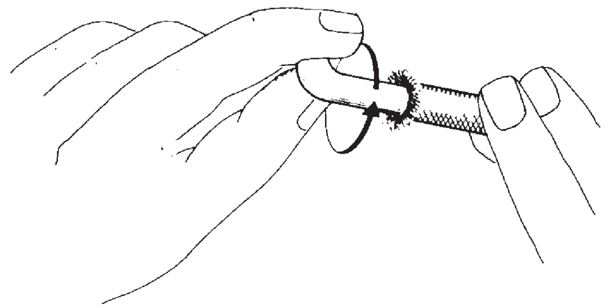
Cutting Insulation
Figure 3



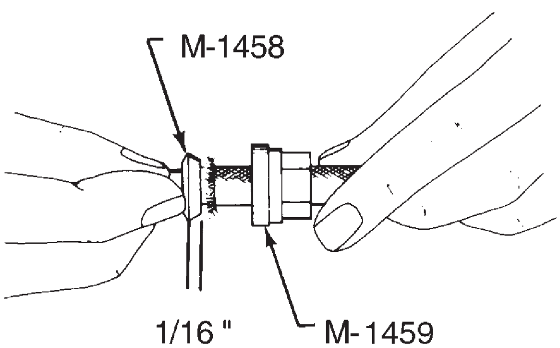
Removing Silicone Rubber from Wire
Figure 4



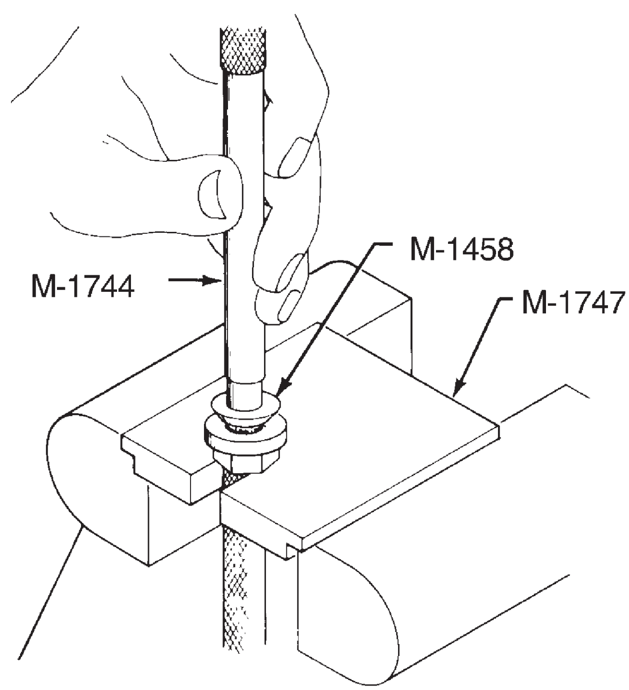
Installation of Plug Endnut
Figure 5



Flaring Out the Shielding
Figure 6



Installation of Ferrule
Figure 7



Driving Tool
Figure 8

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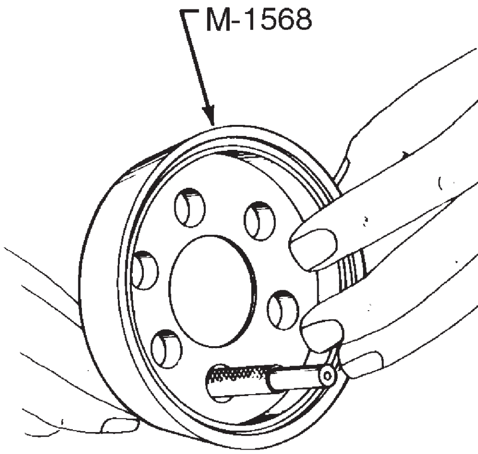
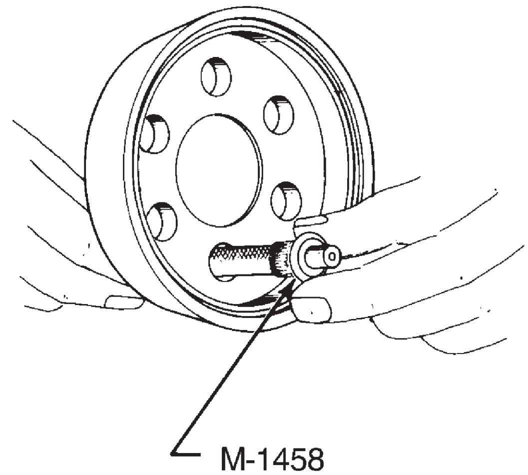
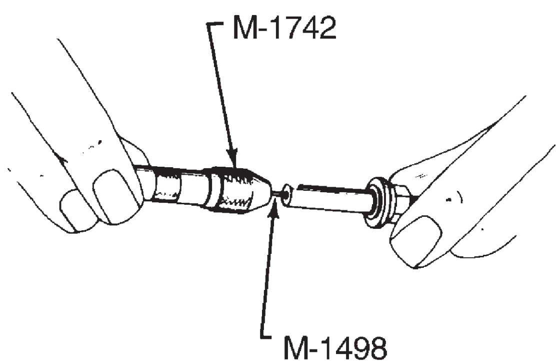


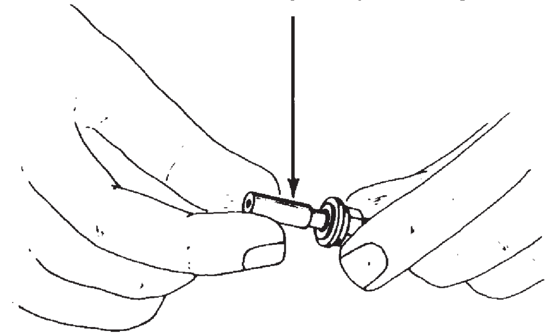
Installation in Housing
Figure 9



Securing Wire in Housing
Figure 10



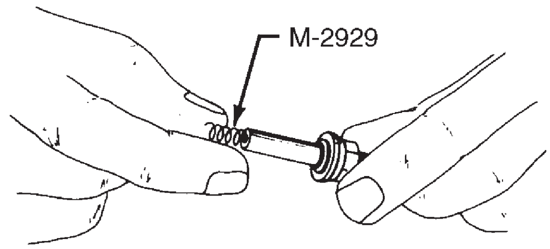
Installation of Electrical Screw
Figure 11



Installation of Insulator Screw
Figure 12

(14) On the magneto end of the wire, place M-3168 Insulator Sleeve over the silicone insulation. On the spark plug end of the wire, use K-3300 Insulator Sleeve Assy (refer to "Figure 12").

(15) Turn M-1455 Spring clockwise on the electrode screw three full turns until the end is flush with the first large coil of the spring. This applies to both ends of the wire (refer to "Figure 13").



Installation of Spring
Figure 13

2. Spark Plugs

A. Removal

(1) Loosen the coupling nut on the harness lead and remove the terminal insulator from the spark plug barrel well. A crows foot adapter is needed to remove the lower spark plugs and special spark plug wrench P/N 8160 for upper plug blocked by the intercoolers (see Note below).

NOTE: The special spark plug wrench P/N 8160 can be obtained from:

Borroughs Tool and Equipment Corp.
2429 North Burdick St.
Kalamazoo, MI 49007-1897

NOTE: When withdrawing the ignition cable lead connection from the plug, care must be taken to pull the lead straight out and in line with the center line of the plug barrel; otherwise, a side load will be applied which frequently results in damage to the barrel insulator and connector. If the lead cannot be removed easily in this manner, the resisting contact between the neoprene collar and the barrel insulator will be broken by a rotary twisting of the collar. Avoid undue distortion of the collar and possible side loading of the barrel insulator.

NOTE: Do not use a torque wrench to remove spark plugs.

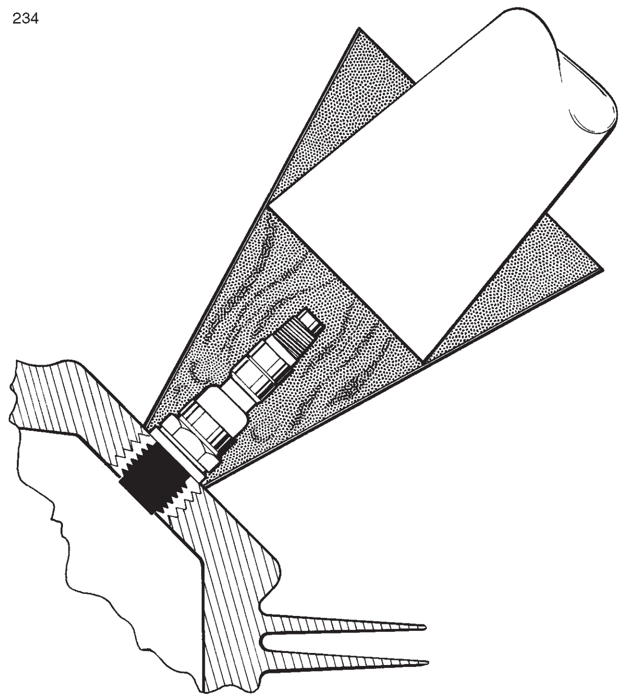
(2) Remove the spark plug from the engine. In the course of engine operation, carbon and other combustion products will be deposited on the end of the spark plug and will penetrate the lower threads to some degree. As a result, greater torque is frequently required for removing a plug than for installing it. Accordingly, the torque limitations given do not apply to plug removal, and sufficient torque must be used to unscrew the plug. The higher torque in removal is not as detrimental as in installation, since it cannot stretch the threaded section. It does, however, impose a shearing load on this section and, if sufficiently severe, could produce a failure in this location.

(3) As soon as they are removed, place spark plugs in a tray that will identify their position in the engine.

NOTE: Spark plugs should not be used if they have been dropped.

(4) Removal of seized spark plugs in the cylinder may be accomplished by application of carbon dioxide by a conical metal funnel adapter with a hole at the apex just large enough to accommodate the funnel of a CO2 bottle (refer to "Figure 14"). When a seized spark plug cannot be removed by normal means, the funnel adapter is placed over and around the spark plug. Place the funnel of the CO2 bottle inside the funnel adapter and release the carbon dioxide to chill and contract the spark plug. Break the spark plug loose with a wrench. A warm cylinder head at the time the carbon dioxide is applied will aid in the removal of an excessively seized plug.

(5) Do not allow foreign objects to enter the spark plug hole.



Removing Frozen Spark Plug
Figure 14

B. Installation

Before installing spark plugs, ascertain that the threads within the cylinder are clean and not damaged.

CAUTION: MAKE CERTAIN THAT THE DEEP SOCKET IS PROPERLY SEATED ON THE SPARK PLUG HEXAGON AS DAMAGE TO THE PLUG COULD RESULT IF THE WRENCH IS COCKED TO ONE SIDE WHEN PRESSURE IS APPLIED.

(1) Apply anti-seize compound sparingly on the threads and install the gasket and spark plugs. Torque the plugs to 420 inch-pounds.

(2) Carefully insert the terminal insulator in the spark plug and tighten the coupling unit.

C. Inspection and Cleaning

(1) Visually inspect each spark plug for the following non-repairable defects:

(a) Severely damaged shell or shield threads nicked up, stripped or cross-threaded.

(b) Badly battered or rounded shell hexagons.

(c) Out-of-round or damaged shielding barrel.

(d) Chipped, cracked or broken ceramic insulator portions.

(e) Badly eroded electrodes worn to approximately 50 percent of original size.

(2) Clean the spark plug as required, removing carbon and foreign deposits.

(3) Set the electrode at 0.016 to 0.021 inches. (See Lycoming Service Instruction No. 1042.)

(4) Test the spark plug, both electrically and for resistance.

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CHAPTER

76

ENGINE CONTROLS

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CHAPTER 76

LIST OF EFFECTIVE PAGES

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CHAPTER 76 - ENGINE CONTROLS

TABLE OF CONTENTS

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POWER CONTROL

1. Engine Setup Procedures

WARNING: BEWARE PROPELLER BLAST AND ROTATING PROPELLER BLADES DURING ENGINE OPERATION. EXERCISE CAUTION TO AVOID DAMAGE TO PERSONNEL AND EQUIPMENT.

CAUTION: BEFORE ATTEMPTING FULL-POWER CHECKS, BE SURE THAT THE BRAKES ARE PROPERLY MAINTAINED AND SET, AND THAT THE GROUND CONDITIONS WILL NOT PERMIT THE WHEELS TO SLIP DURING FULL-POWER CHECK.

2. Idle Speed and Mixture Adjustment

CAUTION: IT IS EXTREMELY IMPORTANT THAT THE ENGINE BE THOROUGHLY WARMED UP. HOWEVER, EXCESSIVE ENGINE TEMPERATURES MUST BE AVOIDED.

NOTE: Allow for the effect of weather conditions and field altitude when making idling adjustment.

A. Start and warm up the engine until the oil temperature reaches 170°F to 190°F.

B. Check magnetos. If the magneto drop is normal, proceed with idle adjustment.

C. Set throttle stop-screw so that the engine idles at 700 rpm. If the rpm changes appreciably after making idle mixture adjustment during the following steps, readjust the idle speed to the desired rpm.

D. When the idling speed has stabilized, move the cockpit mixture control, with a smooth, steady pull, toward the Idle Cut-Off position and observe the tachometer for any change during the leaning process. Caution must be exercised to return the mixture control to the Full Rich position before the rpm drops to a point of engine cut out. An increase of more than 50 rpm while leaning out indicates an excessively rich idle mixture. An immediate decrease in rpm (if not preceded by a momentary increase) indicates the idle mixture is too lean.

E. If step (D) indicates that the idle mixture is too rich or too lean, turn the idle mixture adjustment in the direction required for correction and check this new position by repeating step (D).

F. Make additional adjustments as necessary until a mixture check step (D) results in a momentary pick up of approximately 50 rpm. Each time the adjustment is changed the engine should be run up to 2,000 rpm to clear the engine before proceeding with the rpm check.

G. Make final adjustment of the idle speed adjustment to obtain an idle speed of 700 ± 25 rpm with closed throttle and air conditioner off.

H. The above method aims at a setting that will obtain maximum rpm with minimum manifold pressure. In case the setting does not remain stable, check the idle linkage; any looseness in this linkage would cause erratic idling.

3. Variable Pressure Controller

A. Controller / Throttle Rigging

With the controller lever against the stop in full boost position, adjust the linkage rod to maintain 0.015 / 0.020 inch clearance between the fuel injector throttle lever and the full throttle stop.

See also Lycoming Service Instruction No. 1431, Supplement 2.

B. Adjustment for Manifold Pressure

(1) Warm up the engine until the oil temperature is 170°F to 190°F.
(2) Ensure mixture control is full rich and propeller control is at full rpm (2,500 rpm).
(3) Slowly advance the throttle lever to full throttle or 42.0 in. Hg. manifold pressure.
(4) If the throttle reaches full forward prior to the manifold pressure reaching 42.0 in. Hg., INCREASE manifold pressure by turning the controller adjustment bolt counterclockwise.

NOTE: A general relationship between the number of turns on the controller adjustment and manifold pressure is one turn per 1.0 inch of manifold pressure.

(5) If the manifold pressure reaches 42.0 inches prior to the throttle reaching full forward position, DECREASE manifold pressure by turning the controller adjustment bolt clockwise.
(6) Repeat steps (3), (4), and (5) as required to establish 42.0 in. Hg. manifold pressure at full throttle. Tighten adjustment bolt jam nut.

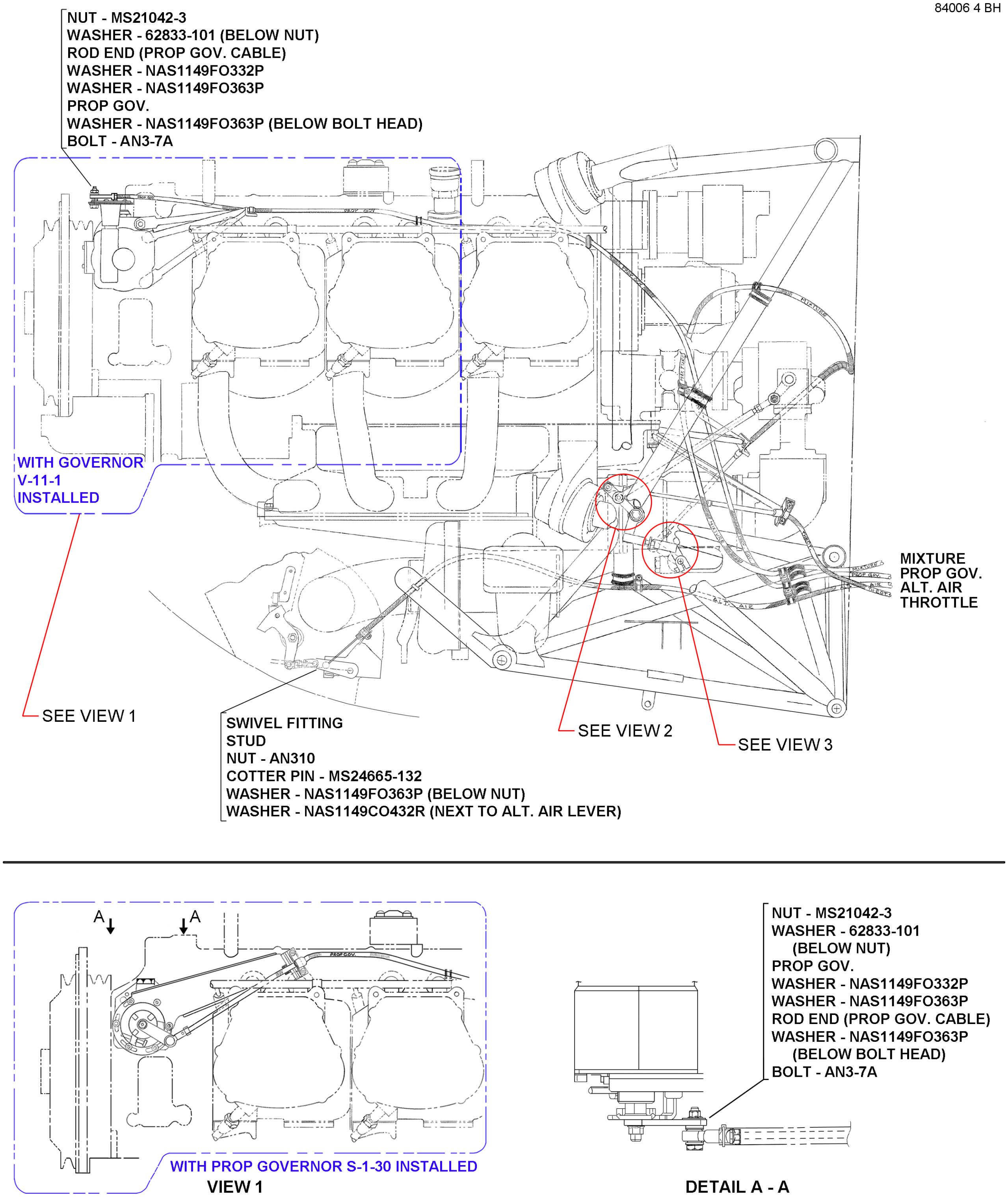
4. Engine Control Cables

A. Installation

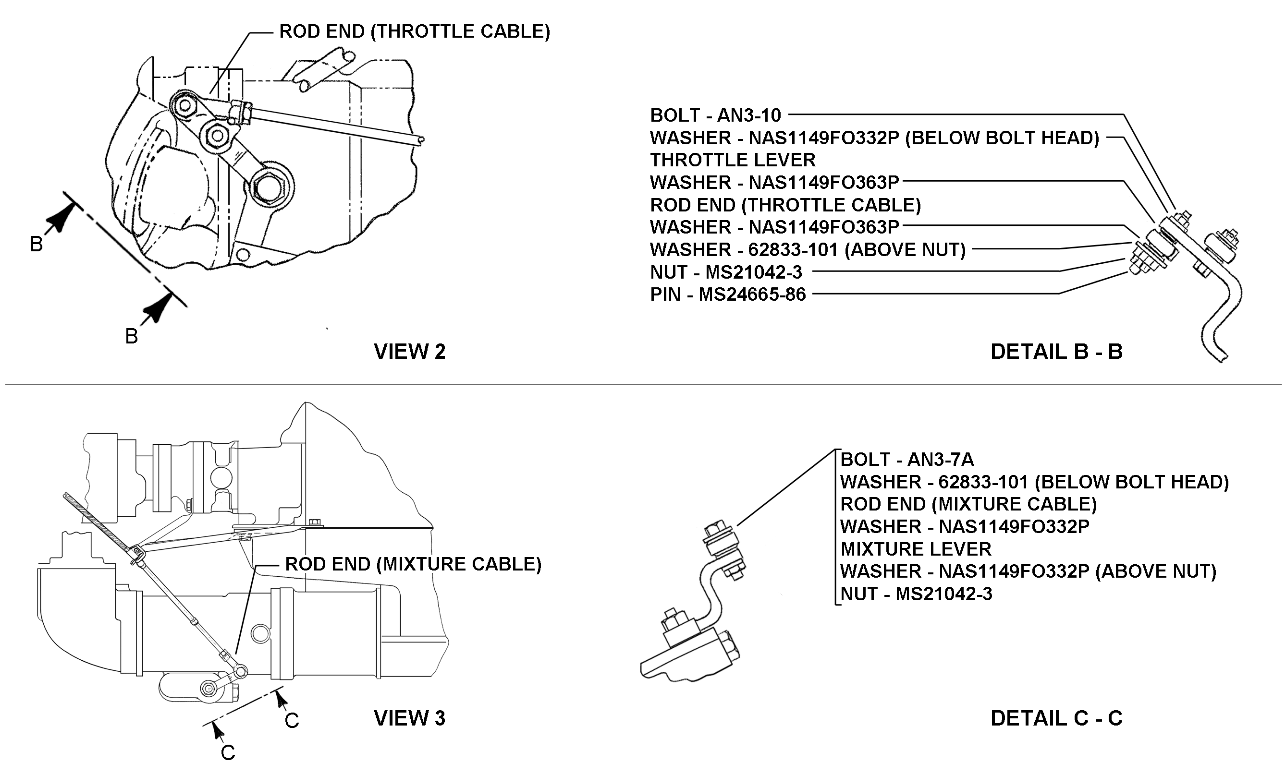
See “Figure 1” for control cable routing and end fitting fastener stackup.

B. Adjustment

(1) With engine throttle control at full throttle, adjust the cable such that the quadrant throttle lever and forward quadrant pin have a .032 inch minimum clearance.
(2) With the prop governor arm at full increase, adjust the cable to provide a .032 inch minimum clearance between quadrant prop lever and forward quadrant pin.
(3) With the engine mixture control at full rich, adjust the quadrant mixture lever to provide a .032 inch minimum clearance from the forward quadrant pin.



Engine Control Cables Installation
Figure 1 (Sheet 1 of 2)



Engine Control Cables Installation
Figure 1 (Sheet 2 of 2)

CHAPTER

77

ENGINE INDICATING

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CHAPTER 77

LIST OF EFFECTIVE PAGES

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CHAPTER 77 - ENGINE INDICATING

TABLE OF CONTENTS

CHAPTER 77 - ENGINE INDICATING

TABLE OF CONTENTS (continued)

POWER

NOTE: In PA-46-350P S/N's 4636375 and up; and PA-46R-350T S/N's 4692001 and up: engine data is collected by the Data Acquisition Unit (DAU) (Avidyne) or Engine/Airframe Unit (GEA) (Garmin G1000/G1000 NXi) and is normally displayed on the Multi-Function Display (MFD). In the event of a MFD failure, all engine instruments can be displayed on the Primary Flight Displays (PFD's). See 34-22-00, 34-25-01 (G1000), or 34-25-02 (G1000 NXi); 77-40-00, and 91-77-40 for more information.

1. Manifold Pressure

A. Manifold Pressure Gauge Installation

Installed in PA-46-350P S/N's S/Ns 4636001–4636021.

(1) Description

The manifold pressure gauge is a vapor-proof, absolute-pressure-type instrument. Pressure from the intake manifold of the engine is transmitted to the instrument through a line. A pointer indicates, in inches of mercury, the manifold pressure available at the engine. A manual drain is provided in the line. The drain line leads from a tee on the back of the manifold air pressure gauge to the drain which is mounted forward of the elevator trim wheel on the left hand side.

(2) Troubleshooting

See "Chart 1".

B. Manifold Pressure Transducer Installation

Installed in PA-46-350P S/N's 4636021 and up and PA-46R-350T S/N's 4692001 and up.

NOTE: PA-46-350P S/Ns 4636021–4636374 incorporate the Transicoil Electronic Module Instrument System (EMIS). See 77-40-00.

The manifold pressure transducer supplies the multi-function display (MFD) or other Integrated Engine Instrument System with mainfold pressure data. The transducer is located in the center and at the top of the forward baggage compartment.

(1) Removal

(a) Remove forward baggage compartment carpet installation.

(b) Remove forward baggage compartment sidewall panels.

(c) Remove forward baggage compartment overhead panel.

CHART 1
TROUBLESHOOTING MANIFOLD PRESSURE GAUGE

Trouble	Cause	Remedy
Excessive error at existing barometric pressure.	Pointer shifted.	Replace instruments.
Excessive error when engine is running.	Line leaking.	Tighten line connections.
Sluggish or jerky pointer movement.	Defective instrument.	Replace instrument.
Dull or discolored marking.	Age.	Replacement instrument.
Incorrect reading.	Moisture or oil in line.	Disconnect lines and blow out.

(d) Disconnect manifold pressure line fitting from forward end of transducer. Disconnect wire connector from aft end of transducer.

(e) Loosen screws holding tube clamps to angle and slide transducer out from clamps.

(2) Installation

(a) Position transducer within tube clamps (2) at the top of the forward baggage compartment. Tighten screws holding clamps to angle.

(b) Connect manifold pressure line fitting at forward end of transducer. Connect wire connector at aft end.

(c) Torque manifold pressure line fitting per label on transducer.

(d) Replace forward baggage compartment overhead panel.

(e) Replace forward baggage compartment sidewall panels.

(f) Replace forward baggage compartment carpet installation.

(3) Test

The manifold pressure transducer (P/N 599-592) can be tested as follows:

(a) Basic

1) Turn aircraft electrical power on to energize cockpit instruments.

2) Prior to engine start, confirm the accuracy of the manifold pressure displayed by the cockpit instrumentation by comparing the indicated manifold pressure against the actual, current barometric pressure, as reported by the local airport authority, or another known good instrument.

3) With a properly functioning Manifold Pressure Transducer, the indicated manifold pressure should agree with the actual barometric pressure within 0.5 inHg (one-half inches of mercury). If an erroneous reading is suspected, continue with detailed test, below.

(b) Detailed

Remove the transducer per Removal, above.

1) Using a calibrated regulated pressure source and a 28 VDC power source, setup the transducer for testing as shown in “Figure 1”.

2) Slowly applying regulated pressure, record the voltage shown on the digital multimeter (DMM) at each stage as shown in “Chart 2”, until a maximum pressure of 25 PSI is reached.

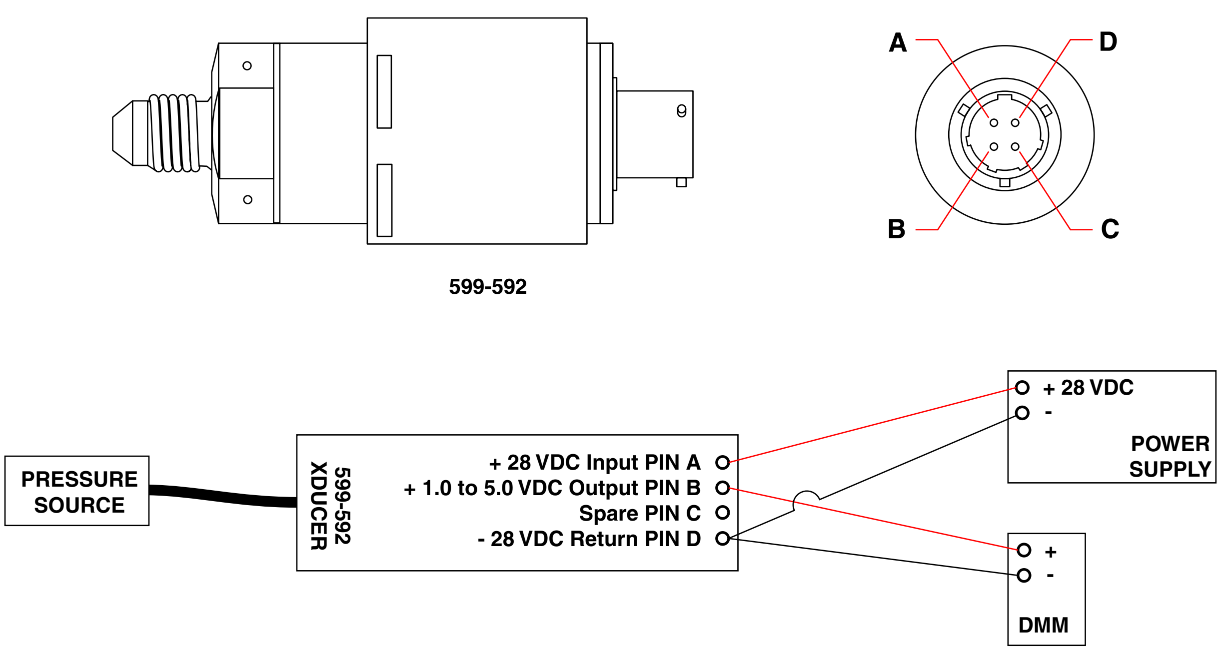
3) Then, slowly decrease the regulated pressure and, again, record the voltage shown on the digital multimeter (DMM) at each stage as shown in “Chart 2”, until zero (0) PSI is achieved.

4) The transducer is good if it meets the specifications shown in “Chart 2”. Any other result is cause for transducer replacement.

5) Install the transducer per Installation, above.

CHART 2
MANIFOLD PRESSURE TRANSDUCER SERVICE TEST SPECIFICATIONS

Pressure Applied (PSI)		P/N 599-592 Nominal ± 2% (VDC)
0		1.00
5		1.80
10		2.60
15		3.40
20		4.20
25	Maximum	5.00
20		4.20
15		3.40
10		2.60
5		1.80
0		1.00



Manifold Pressure Transducer Test Set-up
Figure 1

2. Tachometer Indicator

NOTE: S/N's 4636021 thru 4636374 incorporate Integrated Engine Instrumentation Systems. See 77-40-00.

A. Description

The tachometer is electrically connected to the engine magnetos. The tachometer provides an indication of crankshaft speed in revolutions per minute. The instrument has a recording mechanism for recording the time that the engine is in actual operation. A calibration check on the tachometer is performed every 100 hours. The correct tolerance at 2,500 rpm is (-) 25 rpm (+) 0 rpm.

B. Troubleshooting

See "Chart 3".

CHART 3
TROUBLESHOOTING TACHOMETER INDICATOR

Trouble	Cause	Remedy
No reading on indicator, either permanent or intermittent.	Inoperative instrument. Loose cable connections.	Replace instrument. Tighten cable.
Pointer oscillates excessively.	Inoperative instrument. Excessive friction in instrument.	Replace instrument. Replace instrument.
Indicator changes in climb.	Excessive cleanup in speed cup.	Replace instrument.
Pointer goes all the way to stop, more noticeable in cold weather.	Excessive lubricant in instruments.	Replace instrument.
Pointer jumps at idle.	Speed cup hitting rotating magnet.	Replace instrument.
Tachometer cable breaks.	Cable bent too sharply.	Reroute cable.

TEMPERATURE

NOTE: In PA-46-350P S/N's 4636375 and up; and PA-46R-350T S/N's 4692001 and up: engine data is collected by the Data Acquisition Unit (DAU) (Avidyne) or Engine/Airframe Unit (GEA) (Garmin G1000/G1000 NXi) and is normally displayed on the Multi-Function Display (MFD). In the event of a MFD failure, all engine instruments can be displayed on the Primary Flight Displays (PFD's). See 34-22-00, 34-25-01 (G1000), or 34-25-02 (G1000 NXi); 77-40-00, and 91-77-40 for more information.

1. Turbine Inlet Temperature (T.I.T.)

NOTE: PA-46-350P S/N's 4636021–4636374 incorporate the Transicoil Electronic Module Instrument System (EMIS). See 77-40-00.

A. Description

This instrument is used to monitor the temperature of the exhaust gases as they leave the cylinders (but before they enter the turbocharger turbine).

Although the gauge is adjustable, if it is found defective after checking with the troubleshooting chart, it should be replaced. Replace any faulty leads to the gauge. Because the resistance of the leads is critical for the proper operation of the gauge, use the same type and length of wire as the original when replacing leads.

B. Troubleshooting

See "Chart 1".

C. Gauge and Probe Cleaning and Inspection

CAUTION: DO NOT CONNECT OHMMETER ACROSS METER. IT WILL BURN OUT THE MOVEMENT OF THE METER.

Unless mechanical damage is evident (broken glass, bent or broken pointer, or broken case) the following checks should be performed before removing the instrument:

(1) Remove the engine cowling as necessary to expose the exhaust transition.

(2) Loosen the nut which secures the probe to the waste gate transition assembly. Remove the probe.

(3) Check for a broken weld (at tip end) or a burnt off end. The measured resistance of probe should be 0.73 ± 0.01 ohms. Clean the connections with steel wool.

CHART 1
TROUBLESHOOTING TURBINE INLET TEMPERATURE (T.I.T.) GAUGE

Trouble	Cause	Remedy
Gauge inoperative.	Defective gauge, probe or wiring.	Check probe and lead wires for chafing, breaks or shorting between wires and/or metal structure.
	Adjusting potentiometer turned off scale.	Reset potentiometer.
Fluctuating reading.	Loose, frayed or broken electrical leads or faulty connections.	Clean and tighten connections. Repair or replace defective wire.

(4) Disconnect the lead wires at the instrument and measure the resistance. Resistance with the lead wires connected to probe should be 8 ohms. Clean connections with steel wool.

(5) Make sure that the adjustment screw, located at the rear of the instrument case, is in the center of its travel. If this screw has been turned to either end of full travel, it will shut the instrument off and no indication will be shown on the pointer. With the leads connected to instrument, heat probe using an Alcor testing unit. The meter should read up to the fourth graduation of approximately 1,500° F. If meter still does not read, replace it.

D. Gauge

(1) Removal

(a) Disconnect wires from the gauge at the instrument panel.

(b) Remove four screws which secure the gauge to the instrument panel and remove the gauge.

(c) Remove wires from the wire harness going to the gauge.

(2) Installation

(a) Route the thermocouple wires along with the existing wire harness to the instrument panel.

(b) Install the gauge into the instrument panel and secure it with four screws.

(c) Connect the thermocouple wires to the rear of the gauge.

E. Sensor/Probe Replacement

See “Figure 1”.

NOTE: The normal service life (limit) for the T.I.T. sensor/probe is two hundred-fifty hours time in service.

NOTE: If the sensor/probe time is unknown, initial replacement should occur within the next one hundred hours time in service. If system inaccuracy is suspected, the sensor/probe should be replaced immediately.

(1) Remove the engine cowling as required to expose the exhaust transition.

(2) Remove the existing T.I.T. sensor/probe mounted in a boss welded to the upper portion of the exhaust transition.

(3) Install a new T.I.T. sensor/probe into the hole in the waste gate transition assembly and tighten.

(4) Run the aircraft and operationally check for a proper T.I.T. reading.

(5) Re-install the engine cowling.

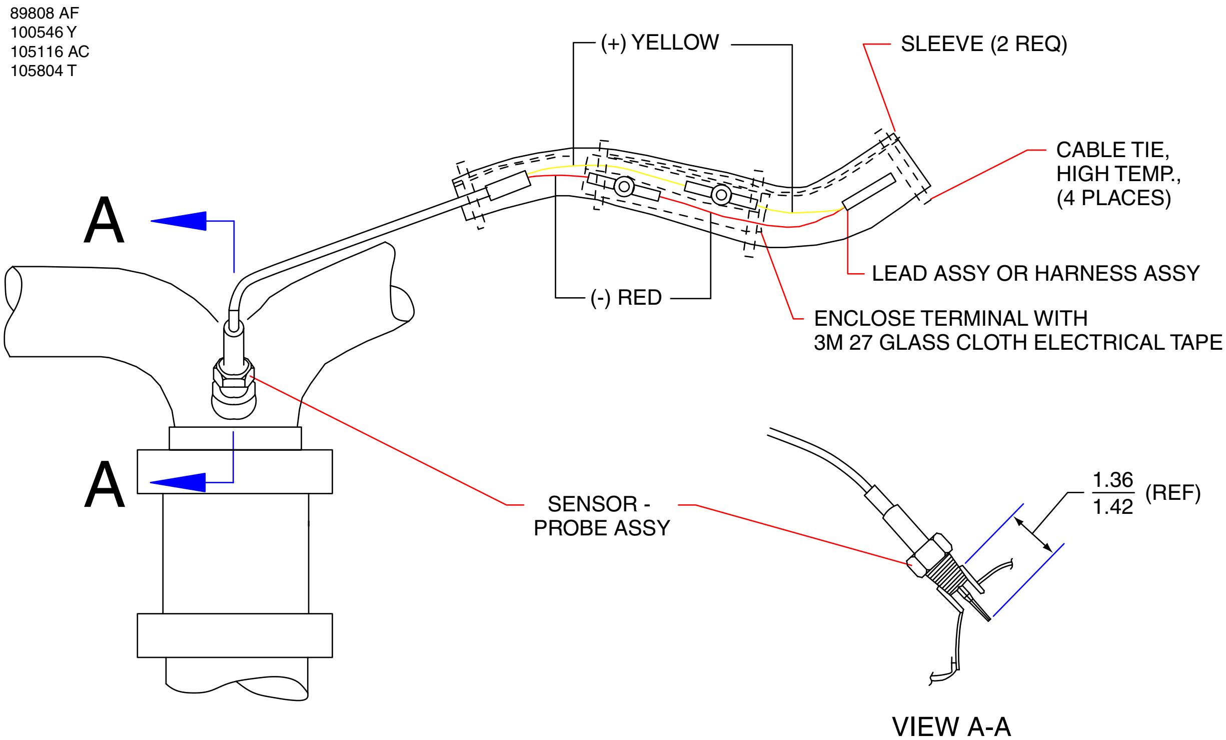
F. System Calibration and Test

This procedure is applicable to the Lewis T.I.T. Indicators, either the standard analog P/N 471-008 or the optional digital P/N 548-811, and is required each time a T.I.T. Indicator is replaced or if system error is suspected. It should be noted that T.I.T. sensor/probe replacement must occur at cylinder changes, engine overhauls or should other T.I.T. system maladies become apparent.

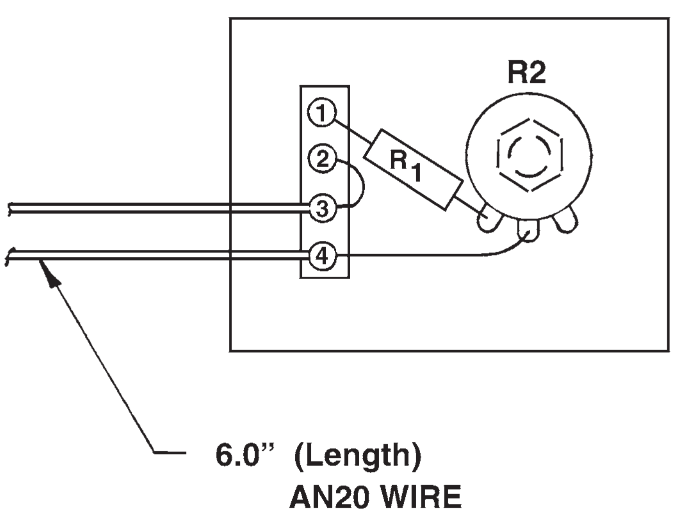
EQUIPMENT REQUIRED: A variable D.C. millivolt power supply and a digital volt-ohm meter. If a millivolt power supply is not available the following items will be required. One 5K Potentiometer JS1N056S502UA (Allen-Bradley part number 481-239) or equivalent. One 27K OHM Resistor ± 10% 1/2 Watt. (See “Figure 2” & “Figure 3”.)

(1) Place the aircraft in an enclosed hangar where the temperature may be stabilized between 50–90° F. (Please take note that the successful completion of this calibration procedure is dependent upon stabilizing the aircraft temperature prior to starting calibration.)

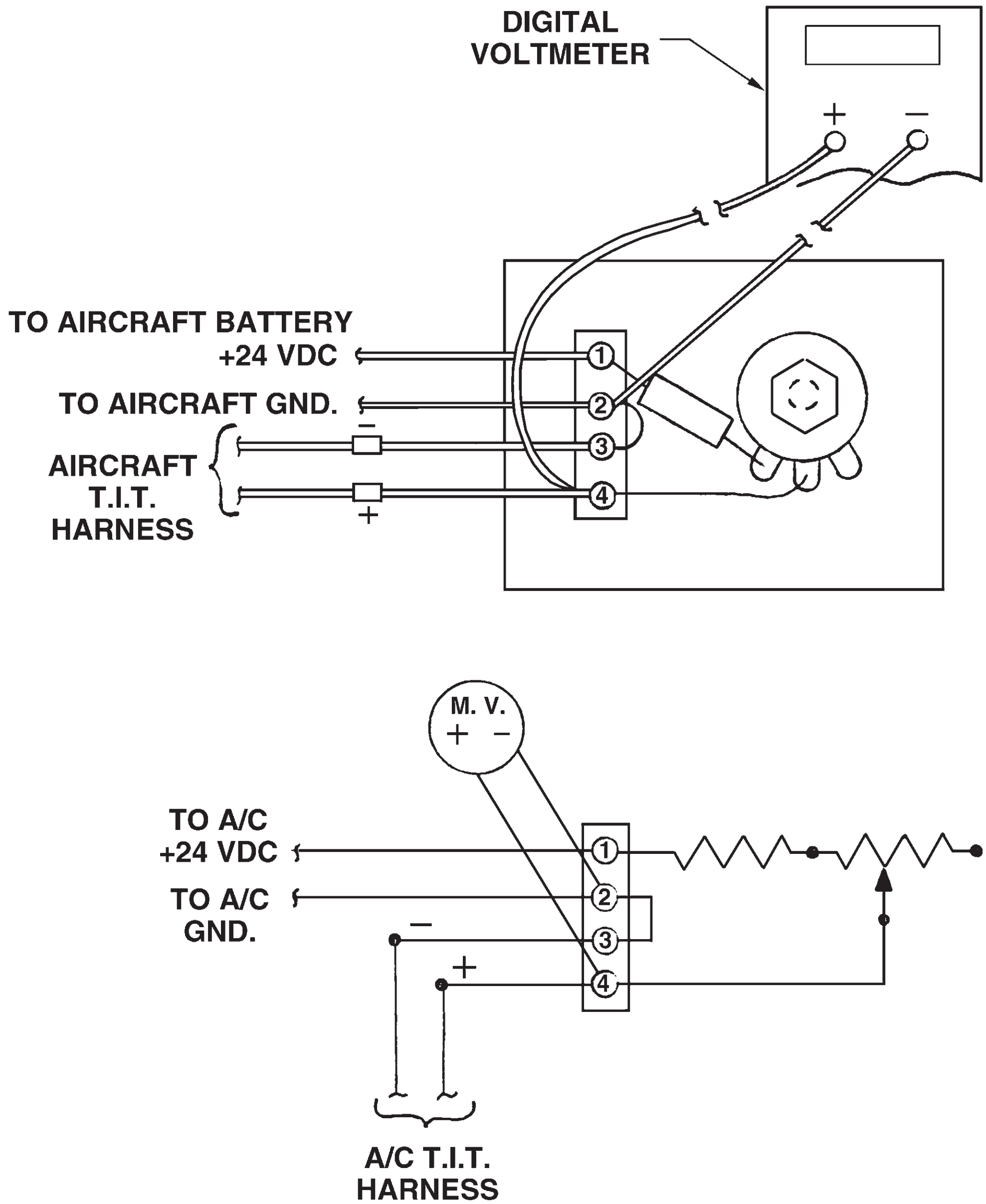
(2) Disconnect the leads of the existing T.I.T. sensor/probe which is mounted in a boss welded to the exhaust transition. (See “Figure 1”.)



T.I.T. Probe Installation
Figure 1



Locally Manufactured Milli-volt Power Supply
Figure 2



TIT System Calibration Set-up
Figure 3

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CAUTION: TO AVOID ERRONEOUS INDICATIONS, DO NOT USE ANY ADDITIONAL WIRING BETWEEN THE MILLIVOLT POWER SUPPLY AND THE T.I.T. WIRING HARNESS EXCEPT AS SHOWN IN “FIGURE 2”.

(3) Connect a stable millivolt power supply directly to the T.I.T. wiring harness. It should be noted that the yellow harness wire is “positive.”

NOTE: If a millivolt power supply is not available, it is permissible to construct the circuit shown in “Figure 2” and connect this circuit to the aircraft battery as shown in “Figure 3”.

(4) Once this calibration configuration is established, allow the aircraft to remain in the hangar for at least 3 hours for the system temperature to stabilize.

CAUTION: DO NOT TOUCH THE CONTACTS ON THE BACK OF THE T.I.T. INDICATOR WITH YOUR HANDS. IF CONTACT IS MADE, ALLOW THE SYSTEM TO STABILIZE FOR AT LEAST ONE HOUR BEFORE MAKING THE ADJUSTMENT TO THE INDICATOR.

(5) Gain access to the back of the T.I.T. indicator. (Remove the glareshield if necessary.)

(6) Adjust the variable D.C. millivolt power supply so the input voltage to the T.I.T. harness is 37.1 ± .1 m.v. It should be noted that this value does not correspond to the standard Alumel-Chromel Tables for “K”- Type thermocouples due to the electrical load of the harness and indicator.

(7) For the standard analog indicator, vary the adjustment screw on the backplate of the indicator until indicator pointer is at the lower edge of the red radial (1750° F.). For the optional analog/ digital indicator, verify that the pointer is at the lower edge of the red radial (1750° F.) - if not, replace indicator.

(8) Re-install the glareshield.

2. Cylinder Head Temperature (CHT)

A. Sensor/Probe Replacement

See “Figure 4”.

(1) Remove the engine cowling as required to expose the cylinders.

(2) Remove the existing C.H.T. sensor/probe mounted in the lower portion of the cylinder.

(3) Install a new C.H.T. sensor/probe into the hole in the cylinder and tighten. Lubricate probe threads and torque as noted in “Figure 4”.

(4) Run the aircraft and operationally check for a proper C.H.T. reading.

(5) Re-install the engine cowling.

B. Combustion Analyzer Gauge

S/N’s 4636001 thru 4636020 only.

(1) Description

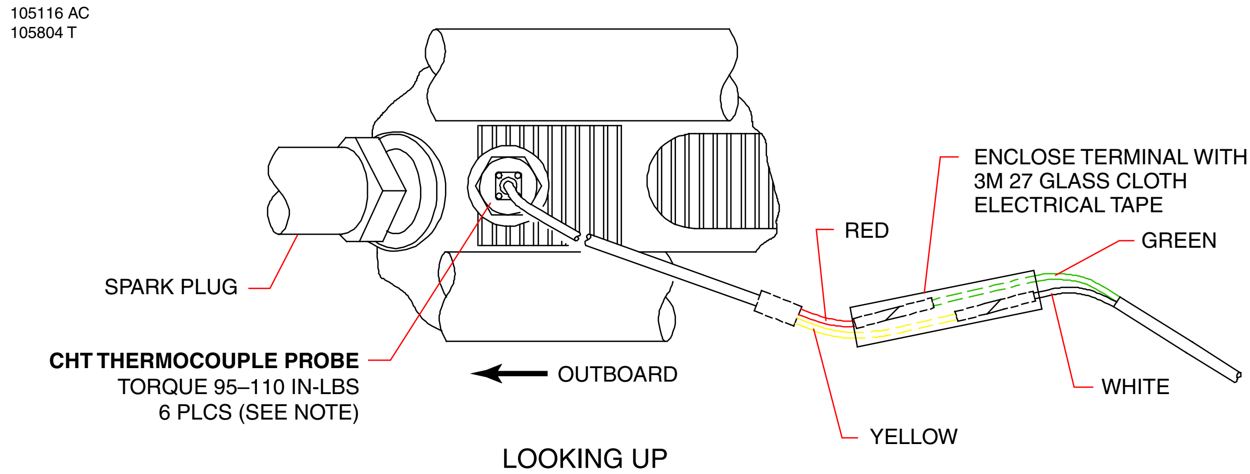
In S/N’s 4636001 thru 4636020 only, a multi-channel combustion analyzer is optional. When installed, it is located on the instrument panel. The combustion analyzer measures the cylinder head temperature in each cylinder head. The combustion analyzer is an electrical instrument and is wired through the instruments circuit breaker.

(2) Troubleshooting

See “Chart 2”.

CHART 2
TROUBLESHOOTING COMBUSTION ANALYZER GAUGE
(S/N's 4636001 thru 4636020 only)

Trouble	Cause	Remedy
75° - 100° rise for one cylinder.	Spark plug not firing due to fouling, faulty plug, lead, or distributor in magneto.	Check spark plug. Clean or replace if necessary. Check lead. Replace if necessary. Check magnetos. Repair if necessary.
75° - 100° rise for ALL cylinders.	One magneto not operating.	Repair faulty magneto.
Increase or decrease, especially after ignition system maintenance.	Improper timing - Increase in EGT means retarded ignition. Decrease means advanced ignition.	Check magneto timing.
Loss of peak EGT.	Poor ignition or vapor in fuel injection system.	Check magnetos. Repair if necessary. Purge fuel injection system if necessary.
Decrease in EGT for ALL cylinders.	Decrease in total airflow carburetor ice or induction ice.	Check alternate air system.
Decrease in EGT for one cylinder.	Intake valve not opening fully - faulty valve lifter.	Check valve lift.
	Scored cylinder or broken ring to cause low compression (EGT may increase due to plug fouling from oil consumption).	Check spark plugs. Clean or replace if necessary. Check compression.
Slow rise in EGT.	Burned exhaust valve.	Check compression.
Decrease in peak and flat.	Detonation - usually the result of 80 octane fuel in 100 octane engine.	Check to make sure proper octane fuel is used.
Sudden off scale rise for any cylinder.	Pre-ignition	During takeoff - Abort if possible. Go to full rich. Check fuel injector flow for proper operation.
Any EGT decrease.	If none of the above causes is evident, suspect a low reading probe or faulty connection.	Check calibration with ALCOR System Tester.
Any increase in EGT.	TROUBLE - because any malfunction of probe, lead, or meter will cause a decrease.	Check probe & leads with ALCOR System Tester. Replace any found to be faulty.



NOTE: Lubricate probe threads with C5-A.

CHT Probe Installation
Figure 4

C. Cylinder Head Temperature (CHT) Scanner

S/N's 4636021 thru 4636374.

The following information describes the Cylinder Head Temperature Scanner used in the Malibu Mirage. This scanner, when connected to the cylinder head temperature gauge and six resistive temperature probes, scans the six probes, determines the hottest cylinder, connects that probe to the gauge and outputs a signal that can be used to indicate which cylinder is the hottest.

(1) Description

(a) Electrical

The Cylinder Head Temperature Scanner gives a discrete output to indicate which cylinder out of six is the hottest. The scanner, momentarily disconnects the CHT gauge and connects all of the probes to measuring circuits that measure the resistance of each probe. The scanner then makes appropriate determinations based upon those measurements, reconnects the gauge to a selected probe, and outputs a discrete output.

An external cycle switch may be used to manually select which cylinder head temperature probe is connected to the gauge. Momentarily grounding the proper pin will step the scanner to Cylinder No. 1. Cylinder No. 1 will be displayed on the gauge for approximately 8 seconds (the “heartbeat” period). Upon the next heartbeat the gauge will again display the hottest cylinder. Again momentarily connecting this pin to ground before the heartbeat will step the scanner to Cylinder No. 2. Now Cylinder No. 2 will be displayed on the gauge for the heartbeat period. Upon the next heartbeat the gauge will again display the hottest cylinder. Continuing to momentarily connect this pin to ground before the heartbeat will step the scanner to the next cylinder in rotation (3, 4, 5, 6, etc.). By maintaining a ground on this pin, any particular cylinder being displayed will continue to be displayed until the ground is removed and the heartbeat cycle occurs. Heartbeat will not occur so long as this pin is grounded.

Six discrete lamps are used to display the hottest cylinder (1 through 6), and are mounted adjacent to the external cycle switch.

An alarm output is also generated when any cylinder is over the preset limit for a “Hot” cylinder. This alarm output is interfaced with the Exceedance Audio Alert Horn. When the external cycle control is in use, the alarm will maintain the last condition displayed when in the automatic mode, regardless of the current condition during the manual external cycling.

(b) Physical

The Cylinder Head Temperature Scanner is a single unit with an integral heat sink and requires no forced cooling. The internal circuits are arranged on two printed circuit boards mounted parallel to the each other. All connections between boards are made by means of two part connectors.

(2) Troubleshooting

In the event of a detectable scanner failure, the scanner will automatically connect the gauge to the probe for cylinder number 3.

D. Cylinder Head Temperature Indicator Calibration Check

(PIR-PPS60182, Rev. A.)

S/N's 4636021 thru 4636374.

All equipment necessary for the proper operation of the CHT, including the CHT/VAC Dual Analog Module Indicator, the Cylinder Head Temperature Scanner, and all six CHT probes with proper harnesses, must be installed in the aircraft and be operational prior to the test, adjustment, and validation described below.

(1) Testing

(a) Remove CHT probe from #1 cylinder.

(b) Disconnect harness from probe for #1 cylinder.

(c) Connect test box (see "Figure 5", construct locally if required) to the two leads removed from probe for #1 cylinder.

(d) Turn switch on test box to 100° F and turn on aircraft power. The CHT indicator should read 100° F ± 10° F. Record displayed value.

NOTE: Engine temperature must be less than 100° F. If temperature is higher than 100° F, press CHT scanner cycle switch repeatedly until lamp for #1 cylinder on instrument panel is illuminated. Hold switch in until temperature is displayed on Quad Digital Indicator (QDI) / Enhanced Digital Indicator (EDI) (see 77-40-00).

(e) Turn switch on test box to 450° F. The CHT indicator should read 450° F ± 10° F. Record displayed value.

NOTE: CHT scanner must be operating during these tests.

(f) If the CHT indicator meets the specified limits, go to Securing the System, below. If the CHT indicator does not meet the specified limits, proceed to Adjustment or Validation, below, depending on aircraft serial number.

(2) Adjustment

P/N 599-588 only - original equipment in S/N's 4636021 thru 4636131 only.

NOTE: P/N's 602-299 or 602-247, installed as original equipment in S/N's 4636132 thru 4636374 are backward compatible and may be encountered in S/N's 4636021 thru 4636131 as service spares. They can be identified by the lack of potentiometers and are not adjustable. If encountered, validate per paragraph C, below.

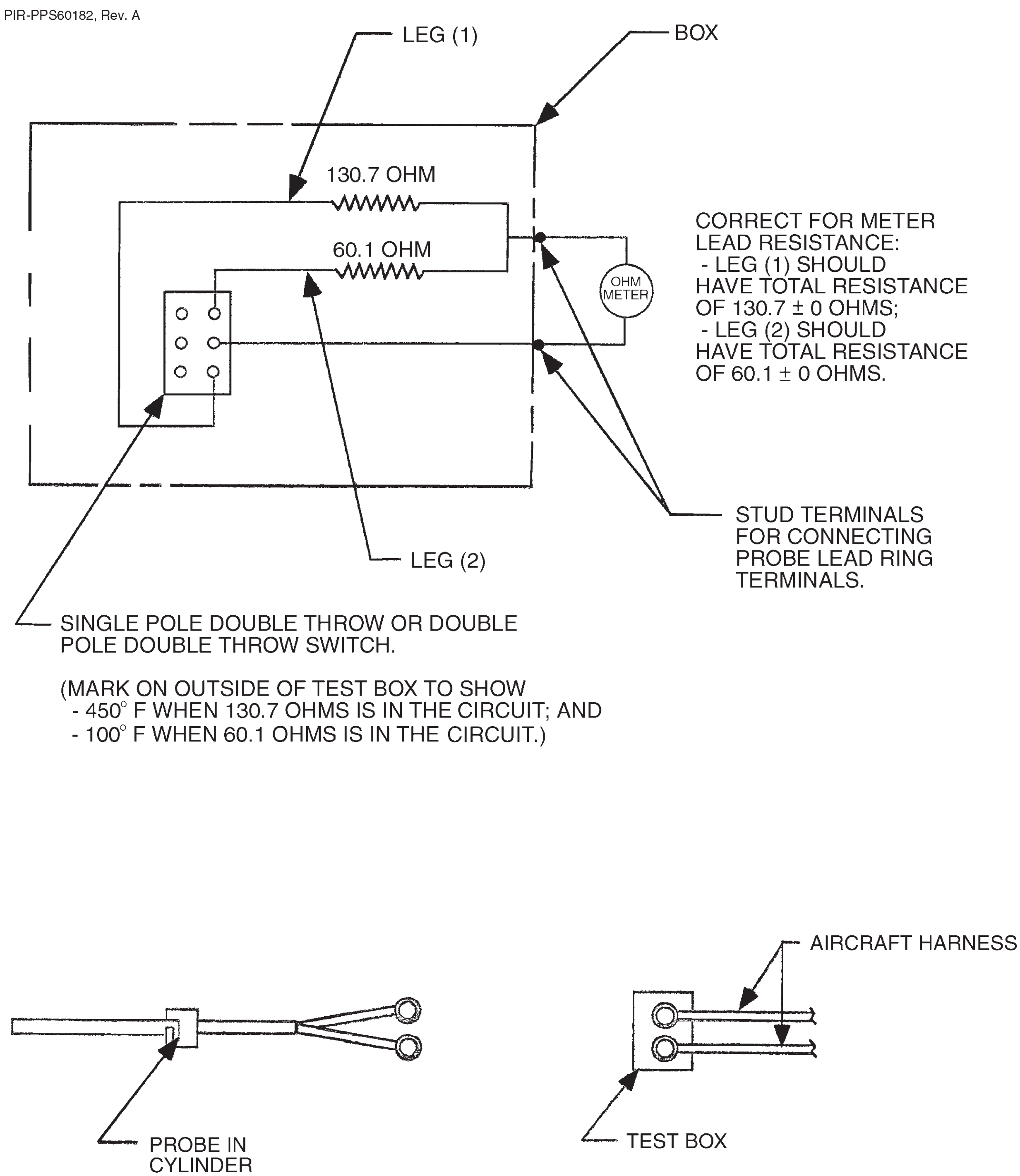
(a) With test box connected per Testing steps (1) - (3), above, select 100° F on test box. Remove CHT/VAC dual analog module indicator from the instrument panel far enough to remove foil sticker from the side of the CHT indicator. Adjust (z) zero potentiometer so that the QDI displays 100° F.

(b) Select 450° F on test box. Adjust (s) span potentiometer so that the QDI displays 450° F.

(c) Return test box switch to 100° F setting and check that CHT indicator reads 100° F ± 10° F. If it does not, readjust the (z) zero potentiometer.

(d) Set switch on test box to 450° F setting and check that CHT indicator reads 450° F ± 10° F. If it does not, readjust the (s) span potentiometer.

(e) Repeat steps (3) and (4) until no adjustment is required to repeat 100° F and 450° F readings when the test box is set to the corresponding switch position.



Effectivity
4636021 thru 4636374

CHT Test Box Construction and Lead Connection
Figure 5

(3) Validation

P/N's 602-299 or 602-247 - original equipment in S/N's 4636132 thru 4636374.

(a) Pull circuit breaker for engine monitor (i.e. - scanner) and repeat test in Testing, above, except test #3 cylinder in lieu of #1.

(b) If CHT indicator fails to meet the specified limits on #3 cylinder also, replace CHT/VAC dual analog module indicator.

(c) If CHT indicator meets the specified limits on #3 cylinder, troubleshoot CHT probe and harness on #1 cylinder and repeat test in Testing, above.

(4) Securing the System

(a) Replace foil sticker over trim potentiometers, if removed.

(b) Reinstall CHT/VAC dual analog module indicator in instrument panel, if removed.

(c) Disconnect and remove test box.

(d) Reconnect harness to probe(s) and reinstall probe(s) in correct cylinder(s).

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INTEGRATED ENGINE INSTRUMENT SYSTEMS

1. Transicoil Electronic Module Instrument System (EMIS)

Installed in PA-46-350P S/N's S/N's 4636021–4636374.

The Transicoil Electronic Module Instrument System (EMIS) is a precision measurement and display system containing both analog and digital displays of engine related parameters (See "Figure 2"). The EMIS is comprised of two parts:

-- Dual Analog Module Indicators, and

-- Quad Digital Indicator (QDI) (S/N's 4636021 thru 4636131), or

Enhanced Digital Indicator (EDI) (S/N's 4636132 thru 4636374) display.

Both Dual Analog Module Indicators and the QDI/EDI are face-mounted instruments and may be removed and installed per 39-10-00.

A. Dual Analog Module Indicator

See "Figure 1" on page 77402 and "Chart 2" on page 77407.

The Dual Analog Module Indicator is a precision micro-processor based instrument for displaying engine parameters. Each module consists of two completely independent analog indicators. Each indicator displays its respective engine parameter reading on the analog dial as well as transmits digital data to the QDI/EDI via the data bus. A rear connector is provided for system interface and analog sensor inputs. Alarm outputs are provided where applicable.

(1) Features

Each Dual Analog Module Indicator features the following (See "Figure 1" on page 77402):

(a) Two parallax free dials, each featuring "Return to Zero" pointers which hold just below "0", or the lowest graduation, when power is removed.

(b) Four status indicator LEDs, two green and two red, with the exception of the Dual Fuel Quantity Module which has green and yellow.

(c) Two front-accessible buttons which provide a simple user interface.

(2) Performance

(a) Stand Alone

1) Power Up Characteristics

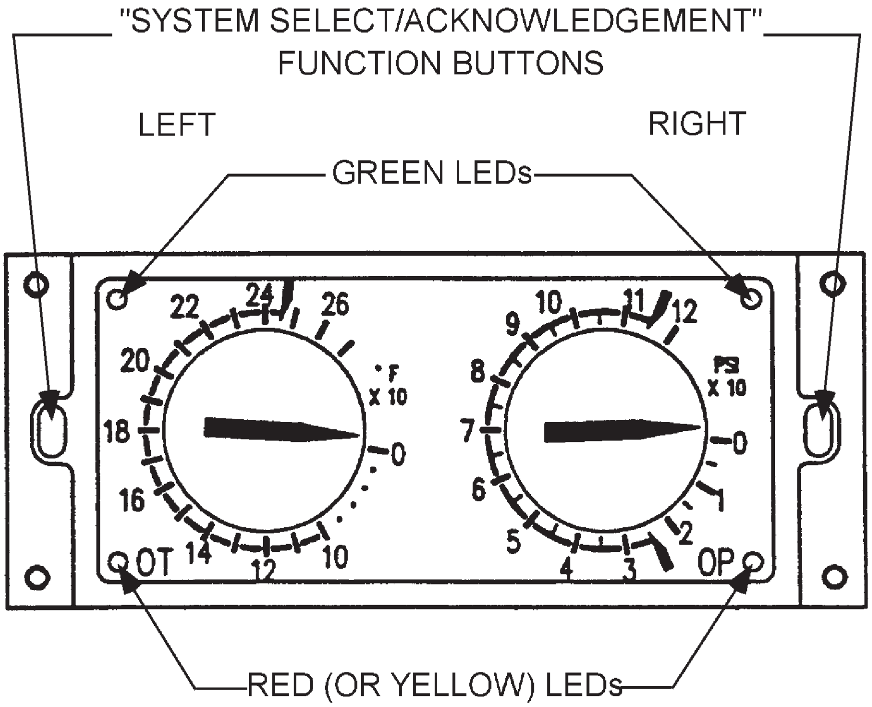
When power is first applied, a stand alone Dual Analog Module will simply drive its pointers towards the correct reading after a short power up delay (less than 0.5 sec).

2) Exceedance Characteristics

The traditional "Red Line", if present, corresponds to a preset limit for that parameter. If the reading exceeds the preset customer specified limit (that may differ from the Red Radial) for more than a specified time delay, the red or yellow LED corresponding to that indicator will illuminate, and the external audible alarm output will turn on.

3) Pressing the switch next to the analog indicator in exceedance will mute the external audible alarm.

NOTE: Additional features, if any, shall be found on the respective analog drawings.



NOTE: Engine OT/OP Dual Analog Indicator is used as an example for this figure.The above dial presentation is for reference only and will vary from one system type to another. Controls and LEDs apply to all analog indicators in the stack. Pointers are shown in power off condition.

Dual Analog Module Indicator (Typical)
Figure 1

(b) System

The Dual Analog Module can be operated as part of a system when a Quad Digital Indicator (QDI)/Enhanced Digital Indicator (EDI) is connected to the data bus.

1) Power Up Characteristics

When power is applied to the system, a seven (7) second power-up self-test is initiated by the QDI/EDI as communication is established between it and the dual analog indicator. The following test functions will occur:

a) Both pointers will drive to the 9 o'clock position.
b) All indicator LEDs illuminate.
c) External audible alarm(s) sound for the last two (2) seconds of self-test.

2) Normal Digital Display Characteristics

Press the select button adjacent to the analog indicator that is to be displayed digitally on the QDI/EDI. When the indicator is being displayed digitally, its green LED will illuminate, indicating that it has been selected.

3) Exceedance Characteristics

The “Red Line” (if present) on the dial face corresponds to a preset limit for that parameter. If the reading exceeds the preset customer specified limit (that may differ from the Red Radial for more than a specified time delay, the red or yellow LED corresponding to that indicator will illuminate and the external audible alarm will sound. The QDI/EDI will automatically select the exceeding indicator to be displayed digitally. This will turn on the green LED for that indicator. The audible alarm will remain on until the reading moves to the normal range, or the select switch next to the parameter in exceedance is pressed. This will mute the external alarm. The red or yellow LED will remain on until the reading returns to the normal range. Once the parameter in exceedance is acknowledged, the exceeding indicator will return to its normal display operation.

4) Retrieval of Exceedances Characteristics

The QDI/EDI stores data concerning an exceedance event which can be retrieved at any time. While the QDI/EDI is displaying this data, both LEDs will flash on the analog indicator corresponding to that data.

5) Self-Test Characteristics

a) Power Up - See Power Up Characteristics, above.

b) Manual - With power applied to the system, press the adjacent select button of the indicator to be tested for two seconds. The indicator’s pointer will drive to 9 o’clock and both LEDs will illuminate. Continue to press button (approx. 5 seconds more) until the external alarm sounds. The audible alarm will remain activated until the select button is released. During this test the digital display will continue to display the actual indicator value.

(3) Troubleshooting

See “Chart 1”.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING DUAL ANALOG MODULE INDICATORS

Trouble	Cause	Remedy
During system self test, analog pointer(s) does not drive to 9 o'clock position but red and green LEDs turn on.	Stuck pointer or faulty drive circuit.	Return module to factory.
During system self test, analog module does not respond.	Loss of power to module.	Check module circuit breaker. Check module connector. Check aircraft wire harness. Return module to factory.
	Loss of communication data bus.	Check module connector. Check aircraft wire harness. Return module to factory.
Analog pointer always drives to the lowest or highest graduation point.	Open or shorted sensor lines.	Check sensor connections.
	Faulty sensor.	Change sensor.
Oil pressure analog instrument goes into exceedance when RPM instrument is below 1400 RPM.	Faulty connection between RPM module connector pin 5 and VAC module connector pin 4.	Check aircraft wire harness.
	Faulty internal circuit in either RPM or OP module(s).	Return RPM and OP instrument modules to the factory.
VAC analog instrument goes into exceedance when RPM is below zero.	Faulty connection between RPM module connector pin 5 and VAC module connector pin 4.	Check aircraft wire harness.
	Faulty internal circuit in either RPM or VAC module(s).	Return RPM and VAC instrument modules to the factory.
Analog instrument cannot be selected by QDI/EDI.	Loss of power to the instrument.	Check module circuit breaker. Check module connector.
	Loss of communication data bus.	Check module connector. Check aircraft wire harness. Reset QDI/EDI. Return module to factory.
	Faulty select switch.	Return module to factory.

CHART 2 (Sheet 2 of 2)
TROUBLESHOOTING DUAL ANALOG MODULE INDICATORS

Trouble	Cause	Remedy
Audio alarm does not function during exceedance.	Loss of connection to alarm.	Check connection from instrument to alarm.
	Faulty alarm.	Change alarm.
	Faulty internal alarm circuit in module.	Return module to factory.
Non-uniform lighting.	Faulty light bulb.	Return module to factory.
Pointer does not return to zero with power-off.	Sticky pointer or faulty return to zero mechanism.	Return module to factory.

B. Quad Digital Indicator (QDI) / Enhanced Digital Indicator (EDI) Module

The Quad Digital Indicator (QDI) / Enhanced Digital Indicator (EDI) is a precision microprocessor based instrument which contains two Dichroic Liquid Crystal Displays (LCDs) with two lines (QDI), or three lines (EDI), each for display, and an annunciator in the lower left corner of each LCD. The LCDs are backlit for use in low ambient light conditions, and contain heaters for low temperature operation. The QDI and EDI are capable of displaying four or six engine parameters at one time, respectively. The QDI lower half is dedicated to engine manifold pressure and RPM while the upper half will digitally display a selected analog indicator reading. The EDI top line is dedicated to engine manifold pressure and RPM, the middle displays selected analog indicators, and the lower displays OAT and % PWR.

Electrical interface is provided by a single 25 pin male “D” subminiature connector. The QDI/EDI contains EMI and RFI protection.

An independent signal conditioning circuit is provided for Outside Air Temperature (OAT).

(1) Functions include:

(a) Automatically posts exceedance warnings.

(b) Stores analog indicator exceedance data for future maintenance review.

(c) Digitally enhances any selected analog indicator in the EMIS system stack.

(d) Performs and posts “Real Time” fuel management calculations.

(e) Automatically posts fuel imbalance using an external panel annunciator.

(2) Front Display and Pushbutton Controls

See “Figure 3” on page 77409.

The QDI/EDI contains two LCDs, one in each window, with two or three lines, respectively, for numerical display. The display is capable of posting four or six parameters at one time. In addition, there are four or six front-accessible buttons to select and/or control the following display modes:

(a) Display Blanking

(b) Fuel Management

(c) Outside Air Temperature

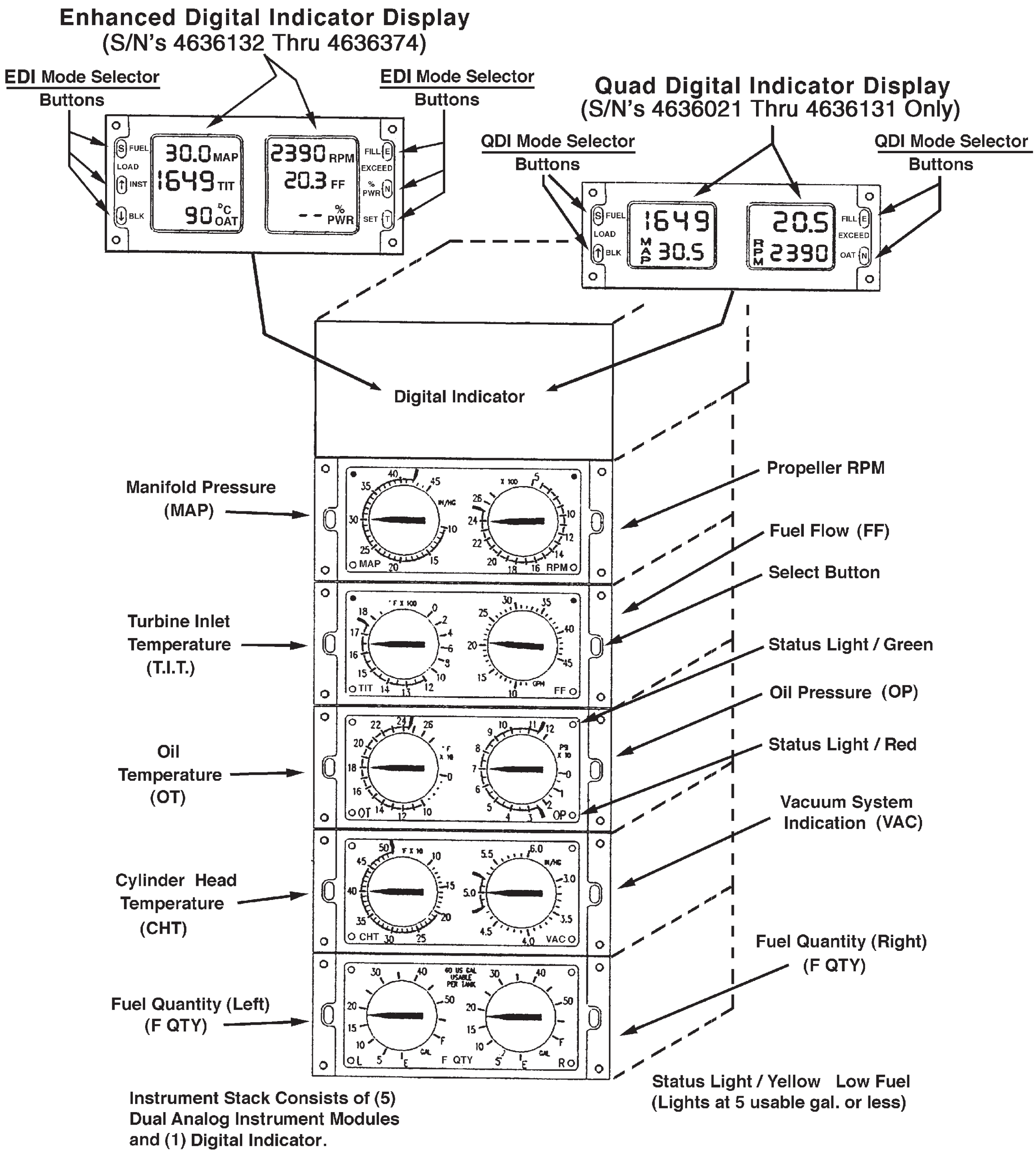
(d) Manual Fuel Load Entry

(e) Maximum Fuel Load Entry

(f) Exceedance Review

(g) Percent Power (EDI only - S/N’s 46360132 thru 4636374)

A summary of these pushbutton functions for the QDI only are listed in “Figure 3” on page 77409. for easy reference. These controls are explained within this section.



Transicoil Electronic Module Instrument System (EMIS)
Figure 2

CHART 2
DUAL ANALOG MODULE INDICATOR SPECIFICATIONS

Indicator	Markings
Manifold Pressure Indicator Housed in the dual manifold pressure/RPM analog module. In addition, the manifold pressure is continuously displayed on the QDI/EDI display.	Range: 10 - 45 in. Hg. Green Arc: 10 - 42 in. Hg. Yellow Arc: None Red Radial: 42 in. Hg
Tachometer Indicator Housed in the dual manifold pressure/RPM analog module. In addition, the RPM is continuously displayed on the QDI/EDI display	Range: 500 - 2700 RPM Green Arc: 600 - 2500 rpm Yellow Arc: None Red Radial: 2500 RPM
Turbine Inlet Temperature Indicator Housed in the dual turbine inlet temperature/fuel flow analog module. Turbine inlet temperature may also be displayed in digital form on the QDI/EDI display by pushing the function switch adjacent to the analog dial.	Range: 0 - 1850° F Green Arc: 1200 - 1750° F Yellow Arc: None Red Radial: 1750° F
Fuel Flow Indicator Housed in the dual turbine inlet temperature/fuel flow analog module. Fuel flow may also be displayed in digital form on the QDI/EDI display (QDI) by pushing the function switch adjacent to the analog dial.	Range: 8 to 45 GPH Green Arc: None Yellow Arc: None Red Radial: None
Oil Temperature Indicator Housed in the dual oil pressure/oil temperature analog module. The oil temperature may also be displayed in digital form on the QDI/EDI display by pushing the function switch adjacent to the analog dial.	Range: 0 - 270° F Green Arc: 100 - 245° F Yellow Arc: None Red Radial: 245° F
Oil Pressure Indicator Housed in the dual oil pressure/oil temperature analog module. The oil temperature may also be displayed in digital form on the QDI/EDI display by pushing the function switch adjacent to the analog dial.	Range: 0 - 120 PSI Green Arc: 55 - 95 PSI Yellow Arc: 25 - 55 and 95 - 115 PSI Red Radial: 25 and 115 PSI
Cylinder Head Temperature Indicator Housed in the dual cylinder head temperature/vacuum pressure analog module. Cylinder head temperature may also be displayed in digital form on the QDI/EDI display by pushing the function switch adjacent to the analog dial.	Range: 100 - 500° F Green Arc: 200 - 500° F Yellow Arc: None Red Radial: 500° F
Vacuum Pressure Indicator Housed in the dual cylinder head temperature/vacuum pressure analog module. Vacuum pressure may also be displayed in digital form on the QDI/EDI display by pushing the function switch adjacent to the analog dial.	Range: 3.0 - 6.0 in.Hg. Green Arc: 4.5 or 4.8 - 5.2 in. Hg. Yellow Arc: None Red Radials: 4.5 or 4.8 and 5.2 in. Hg.
Fuel Quantity Indicator Housed in a dual analog module, left and right fuel tank quantities are indicated independently. Fuel quantity may also be displayed in digital form on the QDI/EDI display by pushing the function switch adjacent to the analog dial.	Range: Empty to Full - 60 Gal. Usable Green Arc: None Yellow Arc: None Red Radial: None

(3) Remote Switch Controls, External Alarm, External Interface Overview

The rear connector provides access to interface:

(a) A remote (normally open) switch to reset the exceedance memory.

(b) An outside air temperature sensor interface.

(c) An open collector output, capable of sinking 2.3 watts max. resistive load, is provided as an output for a fuel imbalance indicator and/or alarm.

(4) Power Up Characteristics

When power is first applied to the EMIS system stack, the following events take place:

(a) First, an automatic system self test is initiated by the QDI/EDI.

1) All analog pointers will be driven clockwise to the 9 o'clock position for approximately seven (7) seconds.

2) All analog indicator LEDs will illuminate.

3) All external alarms will be activated for about 2 seconds at the end of the self test cycle.

4) All active segments of the displays will be energized.

(b) After the self-test is complete, the QDI/EDI steps into its normal digital enhancement mode (see paragraph E.). If the entire EMIS stack is connected, the QDI/EDI will automatically select and display the “MAP”, “RPM”, “TIT” and “FF” indicators. The EDI will also display the “OAT” and, as long as the aircraft is on the ground, a nulled % PWR.

NOTE: Any exceedances that have previously been recorded must be called up manually by the operator.

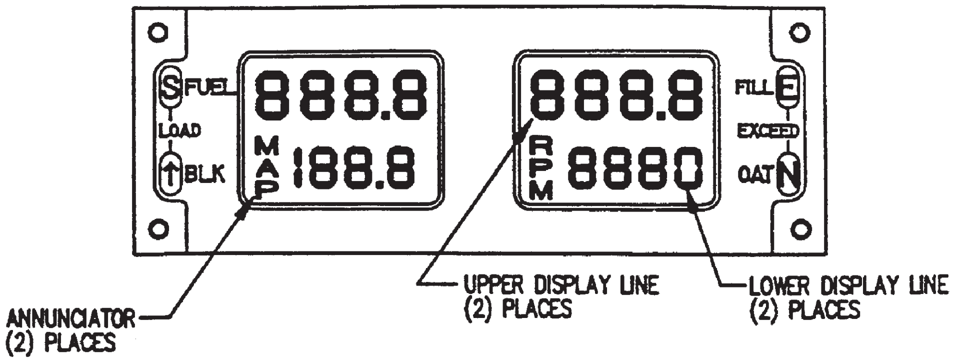
(c) Upon reaching the 9 o'clock self-test pointer position, any analog indicator(s) can be taken out of self-test by depressing and then releasing the select switch associated with the analog indicator(s) you wish to see. The selected indicator(s) will display current sensor readings. The rest of the system stack will complete the self-test cycle.

(5) Normal Digital Enhancement Mode, Analog Indicator

In normal operation mode, the QDI/EDI posts a digital enhancement of the selected analog indicators. When the QDI/EDI is in this mode, four or six parameters are posted simultaneously. In the QDI, two of them are posted in the upper half of the LCDs. The lower half of the display is dedicated to the MAP and RPM analog indicators. (See Figure 4.) In theEDI, two of them are posted in the middle of the LCDs, the top line of the display is reserved for the MAP and RPM analog indicators, and the bottom line displays OAT and % PWR.

To post the corresponding digital reading for an analog indicator, momentarily press the button adjacent to that analog indicator. The selected analog indicator will have its GREEN LED illuminated. The left analog readings will be posted in the upper or middle left LCD and the right analog reading will be posted in the upper or middle right LCD.

NOTE: At power-up the TIT and FF are the default indicators automatically selected for the digital enhancement mode.



NORMAL FLIGHT PUSH BUTTON FUNCTIONS

BUTTON	DISPLAY FUNCTION	FIGURE	SECTION
BLK	BLANK ENTIRE DISPLAY		6.0
FUEL	FUEL MANAGEMENT MODE PRESS BUTTON DISPLAYED INFORMATION once . . . . . . FUEL REMAINING IN GALLONS again . . . . . . FLIGHT TIME REMAINING IN HOURS again . . . . . . FUEL CONSUMED IN GALLONS again . . . . . . RECYCLES DATA	3A 4A 5	7.0
OAT	OUTSIDE AIR TEMPERATURE IN °C	6	9.0

SECONDARY PUSH BUTTON FUNCTIONS SUMMARY

DEPRESS BOTH LEFT BUTTONS "S" AND "↑" TO ENTER
FUEL LOAD ENTRY MODE
(SECTION 8.0)

BUTTON	DISPLAY FUNCTION
S MOMENTARY	SELECTS DIGIT TO BE CHANGED AND SELECTED DIGIT WILL START TO FLASH
↑	INCREMENTS VALUE OF SELECTED DIGIT
E MOMENTARY	AUTO MAX FUEL LOAD 120 GALS AND EXITS TO NORMAL FUEL MANAGEMENT MODE
S PUSH/HOLD (APPROX 2 SEC)	LOADS ENTERED FUEL QTY INTO MEMORY AND EXITS TO FUEL MANAGEMENT MODE

NOTE:
1) FUEL LOAD MEMORY IS NON-VOLATILE.
2) RE-ENTRY WILL POST PRESENT FUEL QUANTITY

DEPRESS BOTH RIGHT BUTTONS TO ENTER
EXCEEDANCE REVIEW MODE
(SECTION 12.0)

BUTTON	DISPLAY FUNCTION	FIGURES
E	PRESS BUTTON DISPLAYED INFORMATION once . . . . . . LAST EXCEEDANCE PEAK VALUE OF FIRST EXCEEDING INDICATOR IN MEMORY again. . . . . . DURATION OF LAST EXCEEDANCE AND AVERAGE VALUE OF EXCEEDANCE FOR ABOVE SELECTED INDICATOR again. . . . . . RECYCLES DATA IF ONLY ONE EXCEEDANCE FOUND IN MEMORY FOR THE SELECTED INDICATOR IN REVIEW - OR - DISPLAYS PREVIOUS PEAK VALUES, AVERAGE VALUES & DURATION IN REVERSE CHRONOLOGICAL ORDER OF SELECTED INDICATOR TO "End ind" PROMPT, THEN REPEATS LIST FROM BEGINNING.	8 10 7B
↑	QUICK ADVANCE THROUGH EXCEEDANCE PEAK VALUES IN REVERSE CHRONOLOGICAL ORDER OF SELECTED INDICATOR PRESENTLY UNDER REVIEW.	
N	SELECTS "NEXT" INDICATOR FOR REVIEW	

"E Clr" MESSAGE = NO EXCEEDANCES IN MEMORY
"End ALL" MESSAGE = ALL EXCEEDANCE EVENTS WERE POSTED

(all display segments as shown are energized during self test)

Quad Digital Indicator Overview
Figure 3

Effectivity
4636021 thru 4636131

(6) Display Blanking

The LCD can be blanked at the operator's discretion. To select this operation, press the "BLK" button. To exit this mode, select any analog indicator from the EMIS system stack or select any other QDI/EDI function.

NOTE: Since the pushbutton designated for blanking the display has secondary functions during exceedance review and fuel management mode, the display blanking function is disabled during these modes of operation.

If an exceedance occurs, the blank mode will end and the exceedance will automatically be posted.

(7) Fuel Management Mode

(a) QDI (S/N's 4636021 thru 4636131 only). The QDI is linked to the fuel flow indicator in the system stack through the data bus. Having this information available allows the QDI to perform "Real Time" fuel management calculations. Fuel remaining, flight time remaining, and fuel consumed can be selected on the display.

To enter the fuel management (posting) mode, momentarily press the "FUEL" button.

1) See Fuel Load Entry Mode, below, for the fuel load entry procedure.

2) If a fuel value had previously been entered into the fuel computer, the fuel remaining will be displayed in gallons. (See "Figure 5" A).

3) To view the flight time remaining, momentarily press the "FUEL" button a second time. (See "Figure 7" A).

NOTE: In the event no fuel flow is detected, the flight time remaining will be displayed as "----". (See "Figure 7" B). When the calculated flight time remaining has expired, "0.0" will be displayed. (See "Figure 7" C).

4) To view the fuel consumed, momentarily press the "FUEL" button a third time. The fuel consumed will be displayed in gallons (See "Figure 8" A). As long as fuel flow is detected, the fuel computer will continue to update the fuel used.

5) Continuing to press the "FUEL" button repeatedly will simply cycle through the three fuel management displays.

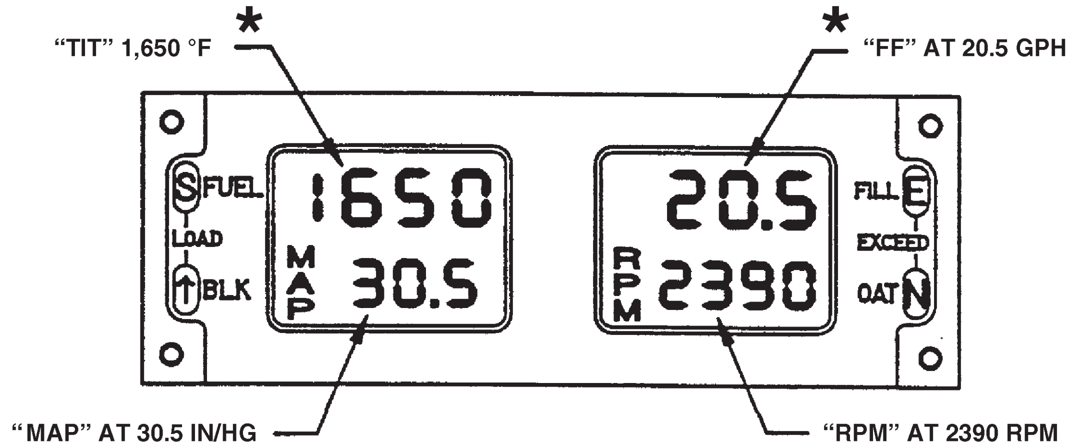
6) To exit the fuel management mode, return to the normal digital enhancement mode and select any analog indicator in the EMIS system stack or any other function of the QDI.

NOTE: The memory allocated for the fuel management mode is non-volatile.

(b) EDI (S/N's 4636132 thru 4636374). The EDI is linked to the fuel flow indicator in the system stack through the data bus. Having this information available allows the EDI to perform "Real Time" fuel management calculations. Fuel remaining, flight time remaining, and fuel consumed can be selected on the display.

1) To enter the fuel management (posting) mode (see "Figure 8" B), momentarily press the "FUEL" button. If a fuel value had previously been entered into the fuel computer, the Fuel Remaining (REM) will be displayed in gallons, as well as: Endurance (ENDUR), Nautical Miles Per Gallon (NMPG), Total Fuel Used (USE), Gallons To Destination (GAL TO DEST) and Gallons At Destination (GAL AT DEST).

2) To exit the fuel management mode and return to the normal digital enhancement mode push the "INST" button.

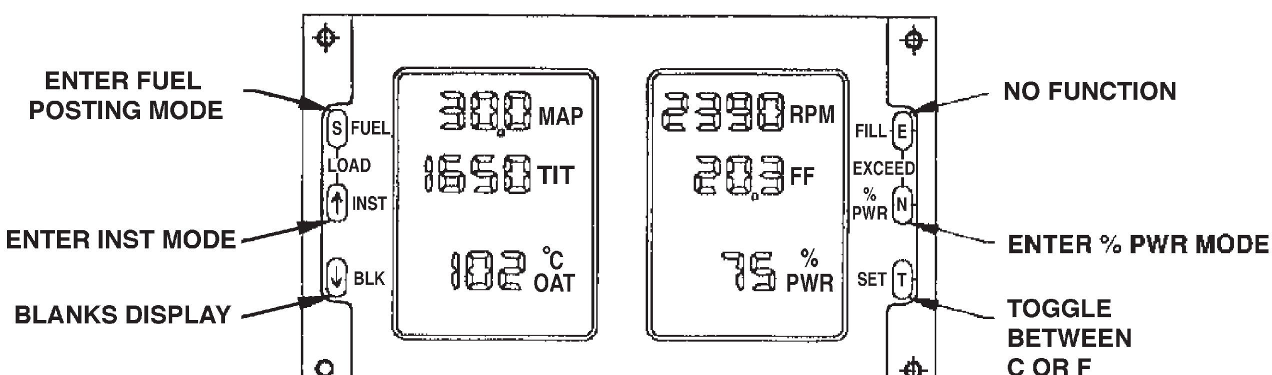


(Lower display lines are dedicated to "MAP" and "RPM" when in Digital Enchancement Mode (i.e. - Normal Display))

* Corresponds to the analog reading of the selected Dual Analog Module Indicator in the EMIS System stack. In this example, the "TIT" and "FF" indicators have been selected and are being displayed digitally.

QUAD DIGITAL INDICATOR

(S/N'S 4636021 THRU 4636131 ONLY)



ENHANCED DIGITAL INDICATOR

(S/N'S 4636132 THRU 4636374)

Digital Enhancement Mode (i.e. - Normal Display)
Figure 4

(8) Fuel Load Entry Mode

(a) QDI (S/N's 4636021 thru 4636131 only).

1) To select fuel load entry mode, press "S" and "UP ARROW" buttons simultaneously.

a) The LCD will display "FUEL 000" or present fuel remaining with the most significant digit found flashing along with a flashing "Pro"gramming prompt in the lower left. (See "Figure 5" B).

b) Pressing the "UP ARROW" button will increment the flashing digit. Pressing the "S"elect button will advance to the next digit. The digits will increment up to nine or the capacity of the aircraft fuel tanks (120 gallons). To enter this fuel load into memory, depress and hold the "S"elect button for approximately 2 seconds. The QDI will automatically enter the normal fuel management mode which will be evident by the absence of the flashing "Pro"gramming prompt and flashing selected digit.

NOTE: The Manual Fuel Load Entry can be re-entered at any time. In this mode, the QDI will post the last fuel remaining (will not update). Depressing and holding the "S"elect button for approximately 2 seconds will:

-- Update fuel load data, if new data is manually entered.

-- Will display updated fuel remaining if no change is made.

2) Max Fuel Load

While in the fuel entry mode, press and release the "E" button with "FILL" adjacent to it. The LCD will display "FUEL 120" and return to the fuel management mode. (See "Figure 5" C).

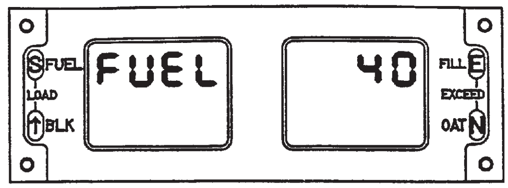
(b) EDI (S/N's 4636132 thru 4636374).

1) Prior to engine start, a new fuel loading can be entered into the EDI by selecting the fuel posting mode. Press the "S" button on the left side of the display. Once in the fuel posting mode, enter the fuel loading mode by simultaneously pressing the "S" and "Up Arrow" buttons ("Figure 5" D and "Figure 5" E). The digital display will show "000" or present fuel remaining with the left most digit flashing. Pressing the "UP ARROW" button will increment the flashing digit up while the "DOWN ARROW" will increment the flashing digit down. Pressing "T" button will toggle between flashing parameter digits. This sequence is repeated until the new fuel loading is displayed in the EDI window. To enter this fuel load into memory, depress the "S" button on the left side of the display. At this point the CDI will return to the fuel posting mode automatically.

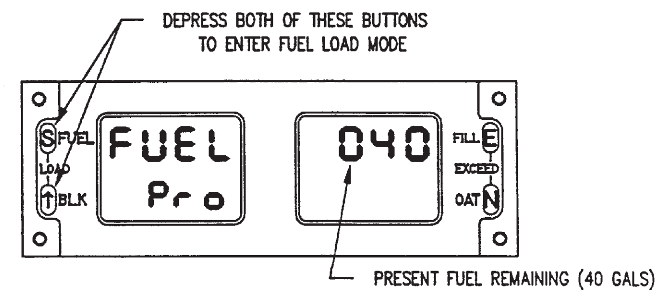
NOTE: During fuel loading entry sequence, the three digits must display 120 gallons or less at all times to permit fuel load entry into the EDI.

2) Max Fuel Load

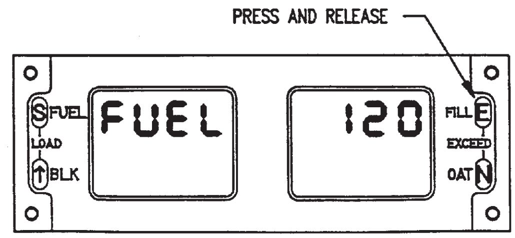
If the fuel tanks are filled to max capacity (120 gallons usable), the pre-set full fuel valve can be entered by depressing the "E" button while in the fuel entry mode ("Figure 5" E).



VIEW A
FUEL DISPLAY SHOWING 40 GALLONS FUEL REMAINING
(FUEL MANAGEMENT MODE)

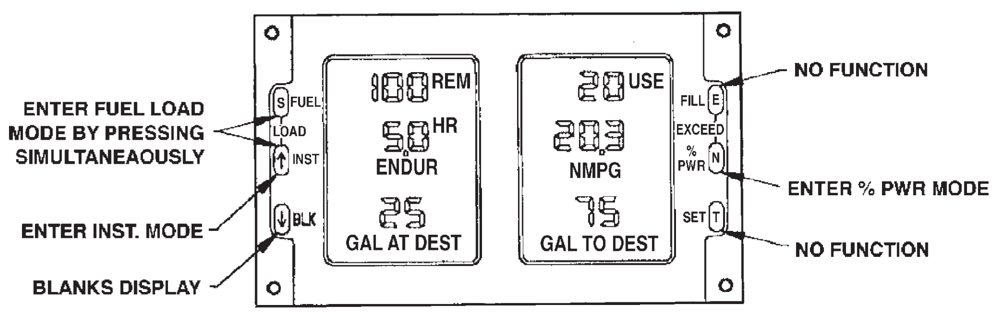


VIEW B
FUEL LOAD ENTRY DISPLAY,
WITH THE "Pro"GRAMMING ANNUNCIATOR
AND SELECTED DIGIT FLASHING

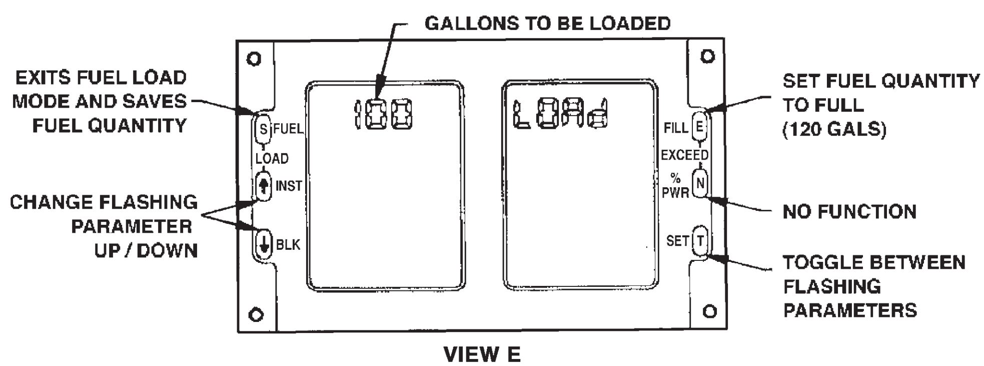


VIEW C
FUEL LOAD DISPLAY
SHOWING MAXIMUM LOAD

QUAD DIGITAL INDICATOR (QDI)
(S/N'S 4636021 THRU 4636131 ONLY)



VIEW D
FUEL MANAGEMENT (POSTING) MODE



VIEW E
FUEL LOAD MODE

ENHANCED DIGITAL INDICATOR (EDI)
(S/N'S 4636132 THRU 4636374)

Fuel Modes
Figure 5

(9) Outside Air Temperature "OAT"

(a) QDI (S/N's 4636021 thru 4636131 only).

1) The "OAT" reading will be posted on the LCD when this function is manually selected.

2) To post the "OAT" reading, press the "N" button with the "OAT" adjacent to it. The "OAT" reading is displayed in degrees C. (See "Figure 6".)

3) To exit from this mode, select any analog indicator from the EMIS system stack, or any other function button of the QDI.

NOTE: Since the pushbutton designated for the "OAT" display has a secondary function during exceedance review mode, the "OAT" display mode is disabled during that mode of operation.

(b) EDI (S/N's 4636132 thru 4636374).

1) "OAT" is the default display in the left LCD when in the INST mode. (See "Figure 4".)

2) When in INST mode, the "OAT" display can be cycled from "° F" to "° C" by pressing the SET button.

(10) Fuel Imbalance Alarm

If a fuel imbalance greater than 10 gallons is sensed between the left and right fuel tanks, the alarm output (for an external annunciator) will be activated after 60 seconds. As soon as the imbalance is corrected, the alarm will deactivate.

(11) Automatic Exceedance Warning Mode

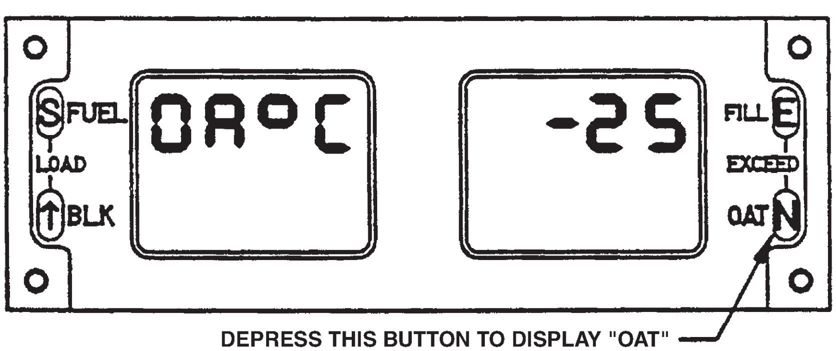
(a) The exceedance warning mode has the highest priority. When an exceedance is detected, the QDI/EDI will automatically select that indicator and display the readings on the LCD. The peak exceedance value of the indicator will flash in the display.

NOTE: The QDI/EDI will automatically select and display exceedances in the order they are detected. Low exceedance recording and the alarm outputs of the VAC and OP analog indicators will be suppressed until the engine reaches a speed of 1400 RPM.

1) When an analog indicator is in exceedance, its red or yellow LED will illuminate and remain on until the unit is no longer in exceedance.

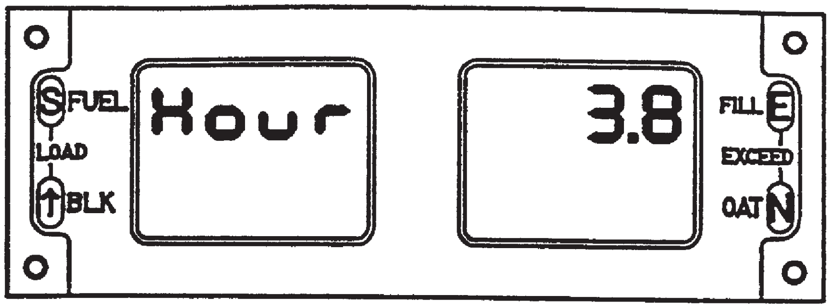
2) The external audible alarm will sound and the LCD display will flash until the operator acknowledges each exceedance by pressing the switch adjacent to the exceeding instrument.

THIS EXAMPLE ILLUSTRATES AN "OAT" READING OF -25°C.

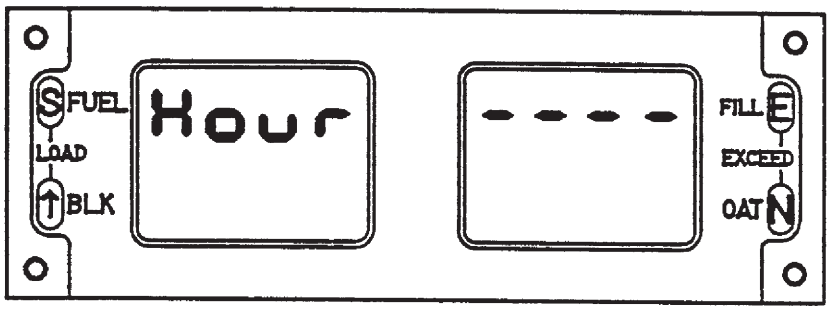


Effectivity
4636021 thru 4636131

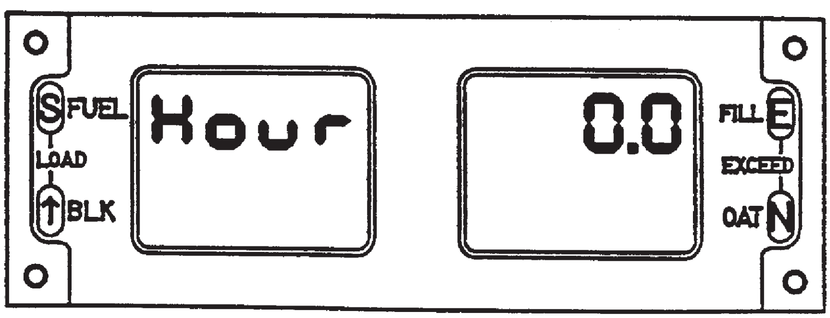
OAT Display - QDI
Figure 6



VIEW A
FLIGHT TIME REMAINING DISPLAY
SHOWING 3.8 HOURS REMAINING
(FUEL FLOW SENSED)



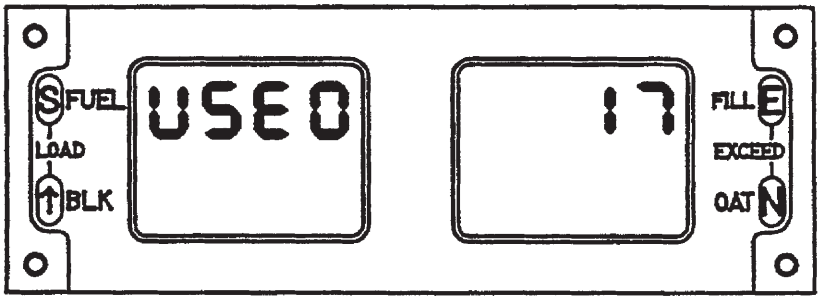
VIEW B
FLIGHT TIME REMAINING DISPLAY
WITH NO FUEL FLOW SENSED



VIEW C
FLIGHT TIME REMAINING DISPLAY
SHOWING FLIGHT TIME EXPIRED

Flight Time Remaining - QDI
Figure 7

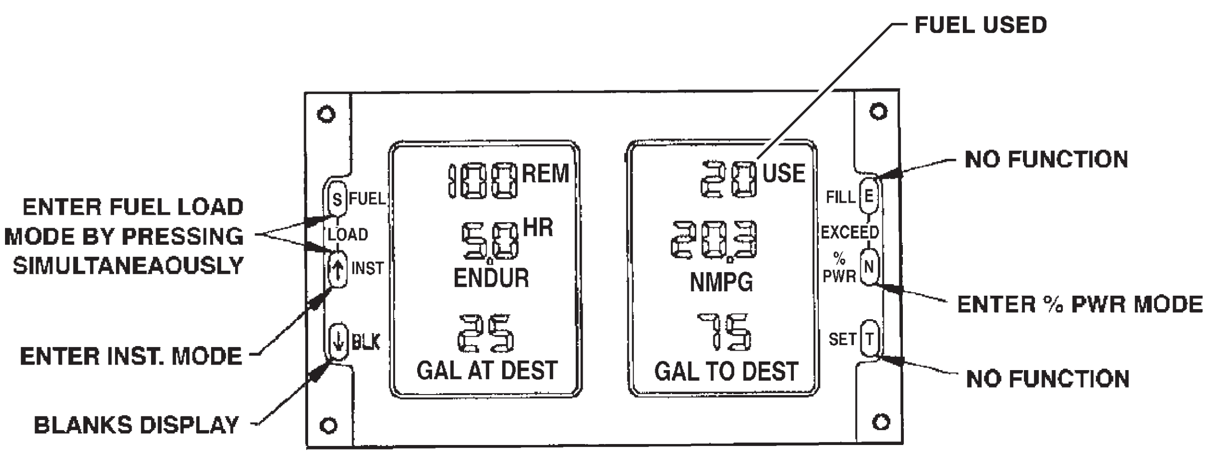
Effectivity
4636021 thru 4636131



VIEW A
FUEL CONSUMED DISPLAY
SHOWING 17 GALLONS OF FUEL USED

NOTE: The fuel flow indicator must be connected to the data bus for input to the fuel computer.

QUAD DIGITAL INDICATOR (QDI)
(S/N'S 4636021 THRU 4636131 ONLY)



VIEW B
FUEL MANAGEMENT (POSTING) MODE
SHOWING 20 GALLONS OF FUEL USED

ENHANCED DIGITAL INDICATOR (EDI)
(S/N'S 4636132 THRU 4636374)

Fuel Used
Figure 8

(b) During the entirety of the exceedance (up to 10 minutes or 599.9 seconds), the QDI/EDI will record the peak value of the exceedance, the average value of the exceedance, and the time duration of the exceedance. This data is stored in non-volatile memory for future recall.

(c) If more than one exceedance occurs at the same time, all will be recorded in the non-volatile memory with only the last exceedance being displayed. The operator must acknowledge each exceedance to mute the external audible alarm. Acknowledgment of each exceedance can be random.

(12) Exceedance Review Mode

(a) The QDI/EDI contains non-volatile memory for future examination of recorded exceedances. The memory size is sufficient to record a total of 238 exceedance events. Each event contains the Peak, Average and Time duration of exceedance. The exceedance history is presented in reverse chronological order for each indicator during review, with the most recent exceedance event listed first.

CAUTION: WHEN THE MEMORY ALLOCATED FOR EXCEEDANCE RECORDING IS FULL, NO NEW EXCEEDANCES WILL BE RECORDED. IT WILL BE NECESSARY FOR THE OPERATOR TO HAVE A QUALIFIED MECHANIC PERIODICALLY RESET THIS MEMORY. THIS MEMORY CAN BE RESET BY USE OF THE REMOTE SECURITY RESET SWITCH. REFER TO PARAGRAPHS C(1), ABOVE, AND M, BELOW.

(b) To enter the exceedance review mode, depress both right hand buttons (with the word “EXCEED” between them) on the front of the QDI/EDI. The operator can exit this mode at any time by selecting any analog indicator.

(c) Display Messages

The QDI/EDI will post four possible messages during the exceedance review operation.

1) “E Clr” (See “Figure 9”, A)

This message indicates that no exceedances were found in the exceedance memory.

2) “END ind” (See “Figure 9”, B)

This message indicates that all of the exceedance events for the indicator under examination have been displayed.

3) “END ALL” (See “Figure 9”, C) This message indicates that all of the exceedance events for all indicators have been displayed.

4) “END ind” and “E FUL” (See “Figure 9”, D)

These messages indicate that all of the exceedance events for the indicator under examination have been displayed and exceedance memory is full.

(d) Pushbutton Functions

1) QDI (S/N's 4636021 thru 4636131).

a) The "E"xceedance Button

This button will cause the QDI to display all the exceedance events for the indicator under examination. Each time this button is pressed, the displayed data will change from the "Exceedance Peak" to the "Average and Duration of Exceedance" for each event in memory until the "END ind" message is displayed. Continuing to press the button will repeat the list.

b) The "UP ARROW" Button

This button will display only the "Exceedance Peaks". The "END ind" message is posted to signify that all peak events have been displayed. Continuing to press this button will repeat the list.

c) The "N"ext Button

This button allows the operator to manually select another indicator to be reviewed. This option is convenient when only the last exceedance peaks for all of the exceeding indicators in memory are required to be reviewed. The "END ALL" message will be posted when all the indicators have been displayed. Continuing to press this button will repeat the list.

2) EDI (S/N's 4636132 thru 4636374).

a) The push button functions used to review the limitation exceedance events in the exceedance review mode "PEAK DISPLAY" ("Figure 10") are:

Down Arrow - This button will cause the EDI display to post all the exceedance events for the indicator under examination. Each time the down arrow button is pressed, the posted data will change to the next peak exceedance for the same indicator.

"N" (Next) - This button allows the operator to manually select another indicator to be reviewed. The "END ALL" message will be posted when all the indicators have been posted. Further depressions will repeat the list.

"E" (Exceedance) - This button allows the operator to enter the exceedance review mode average display.

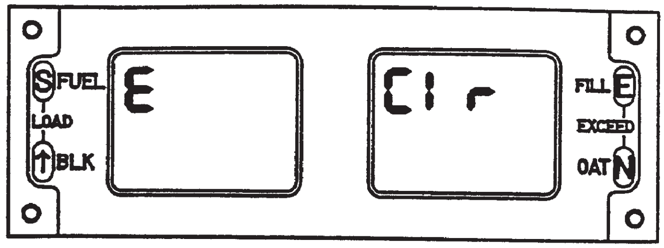
b) The push button functions used to review the limitation exceedance events in the exceedance review mode "Average Display" are: (See "Figure 11".)

"E" (Exceedance) - This button will cause the EDI display to post all the average exceedance events for the indicator under examination. Each time "E" is pressed, the posted data will change to the next average exceedance for the same indicator.

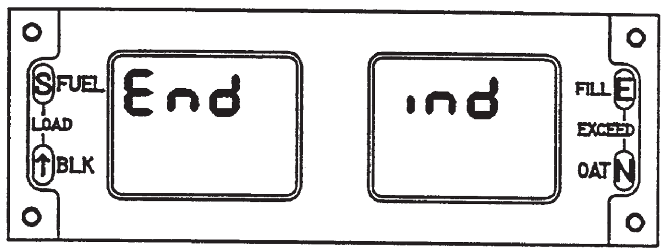
"N" (Next) - This button allows the operator to manually select another indicator to be reviewed. The "END ALL" message will be posted when all the indicators have been posted. Further depressions will repeat the list.

Up Arrow - This button allows the operator to enter the instrument mode display.

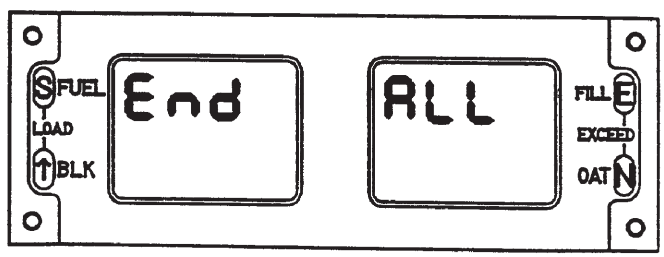
NOTE: QDI shown, EDI similar.



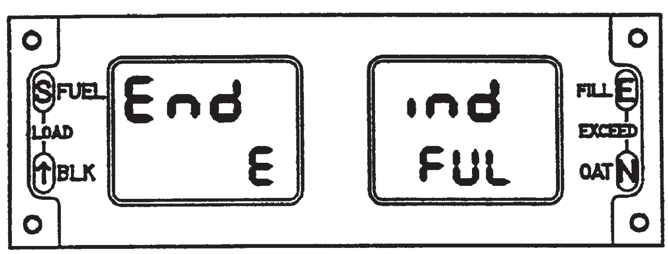
VIEW A
EXCEEDANCES CLEAR MESSAGE



VIEW B
END INDICATOR MESSAGE



VIEW C
END ALL MESSAGE



VIEW D
END INDICATOR AND
EXCEEDANCES FULL MESSAGES

Exceedance Review
Figure 9

(e) Upon entry into the exceedance review mode, two possible presentations will be displayed based on the status of the QDI/EDI's exceedance memory.

1) No Exceedance Data Found In Memory

In the event no exceedance data is found, an "E Clr" prompt will be displayed. (See "Figure 9", A). At this point, the operator can exit the exceedance review mode.

2) Exceedance Data Found In Memory

When exceedance data is found in memory, the QDI/EDI will display the peak exceedance data for the first indicator found in memory.

(f) Peak Exceedance Display

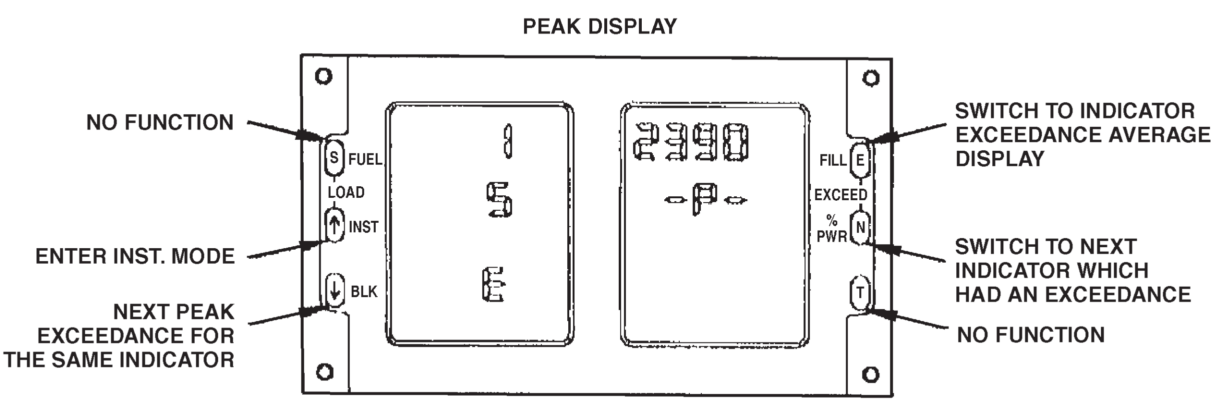
1) The peak exceedance display format for an analog indicator is illustrated in Figure 10. An analog indicator with an exceedance displayed is identified by both the red or yellow and green LEDs flashing. The operation of the analog indicator is not affected during exceedance review.

2) The peak value of the exceedance will be displayed digitally above the analog indicator under review.

3) The opposite display window will contain an "E" and two numbers. The top number represents the exceedance event being reviewed for that indicator. The lower number represents the number of exceedance events that are in memory for that indicator.

(g) Exceedance Average and Time Display

One side of the display will post the average exceedance value in memory, above the prompt "-A-". The other side of the display will post the exceedance time in seconds above the prompt "SEC". (See "Figure 11".)



Peek Exceedance Display
Figure 10

(13) Remote Exceedance Memory Reset Control

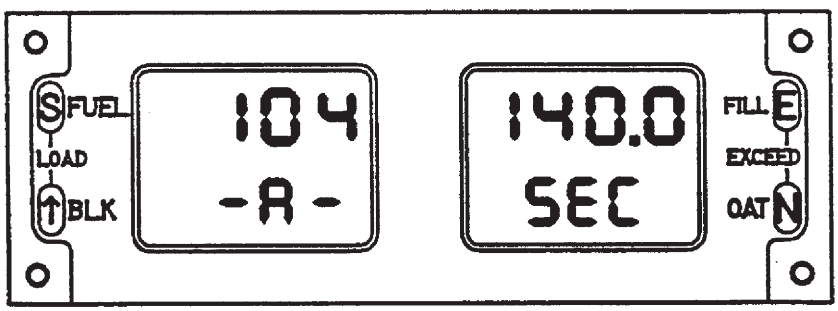
WARNING: ALL EXCEEDANCE HISTORY DATA WILL BE LOST DURING THIS RESET OPERATION, AND CANNOT BE RETRIEVED.

The QDI/EDI is configured to accept a remote exceedance memory reset control (See Remote Switch Controls, External Alarm, External Interface Overview; above). Upon receipt of the control signal, the QDI/EDI will reset (clear) its NVRAM exceedance memory. This control can be initiated at any time.

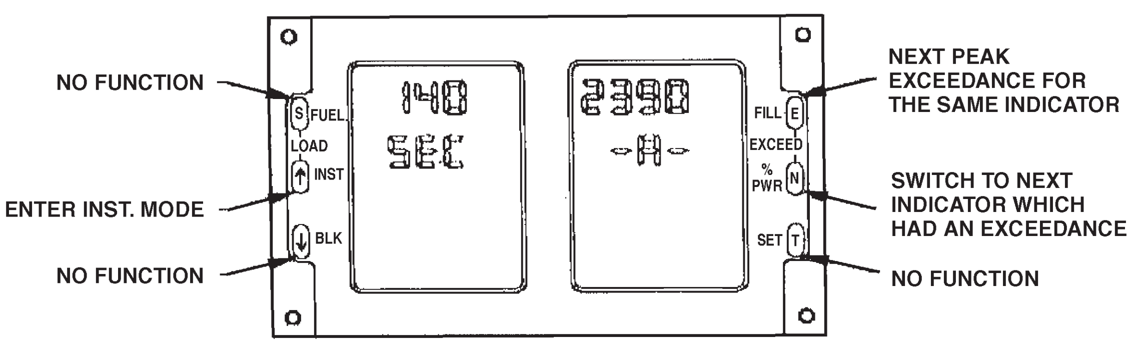
(14) Servicing

The Quad Digital Indicator/Enhanced Digital Indicator cannot be serviced in the field. Should any of the preceding checks indicate a failure, return the unit to the factory for service.

AVERAGE EXCEEDANCE VALUE AND EXCEEDANCE TIME IN SECONDS MESSAGES



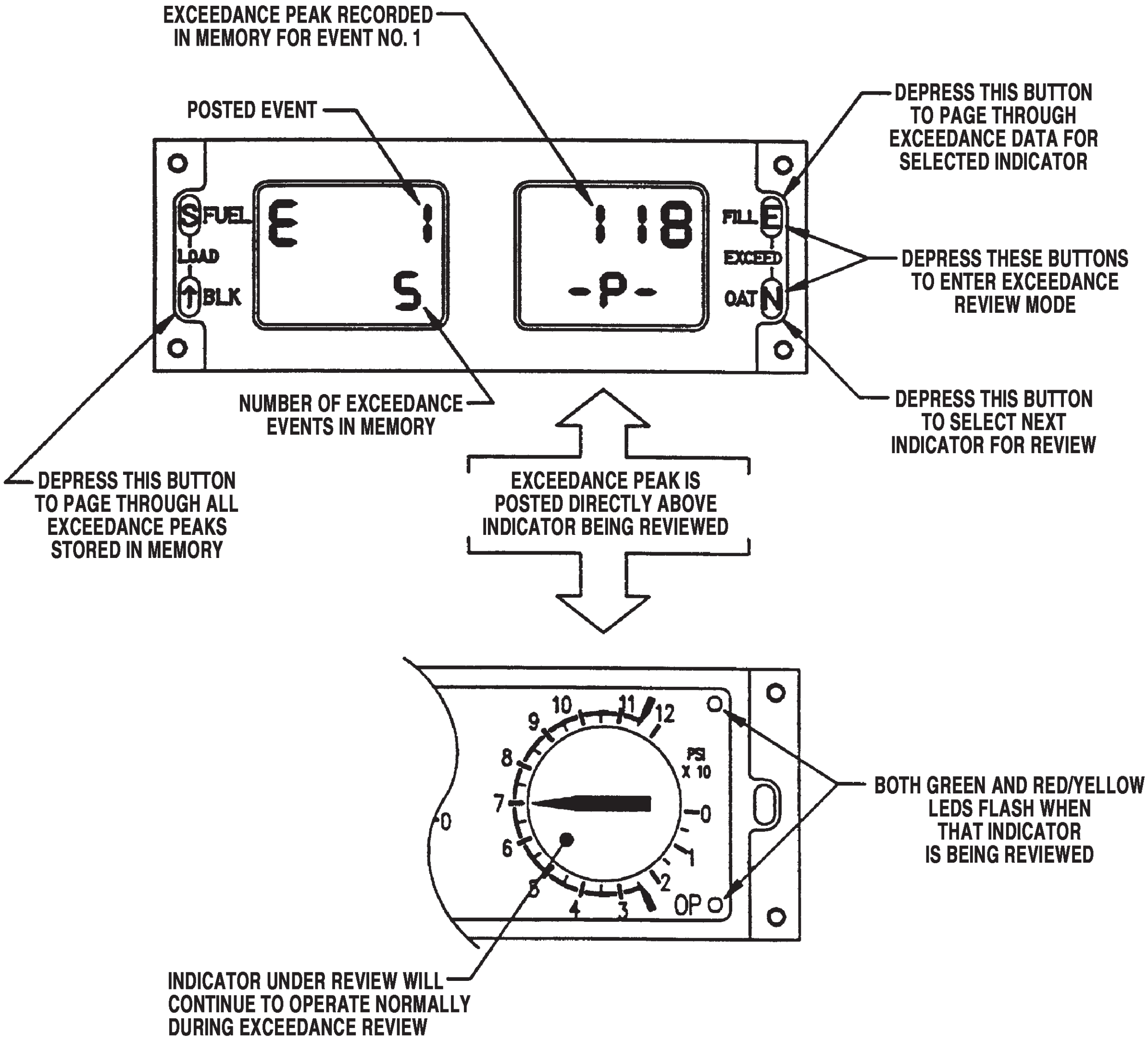
QUAD DIGITAL INDICATOR (QDI)
(S/N'S 4636021 THRU 4636131 ONLY)



ENHANCED DIGITAL INDICATOR (EDI)
(S/N'S 4636132 THRU 4636374)

Exceedance Average and Time Display
Figure 11

QDI SHOWN, EDI SIMILAR

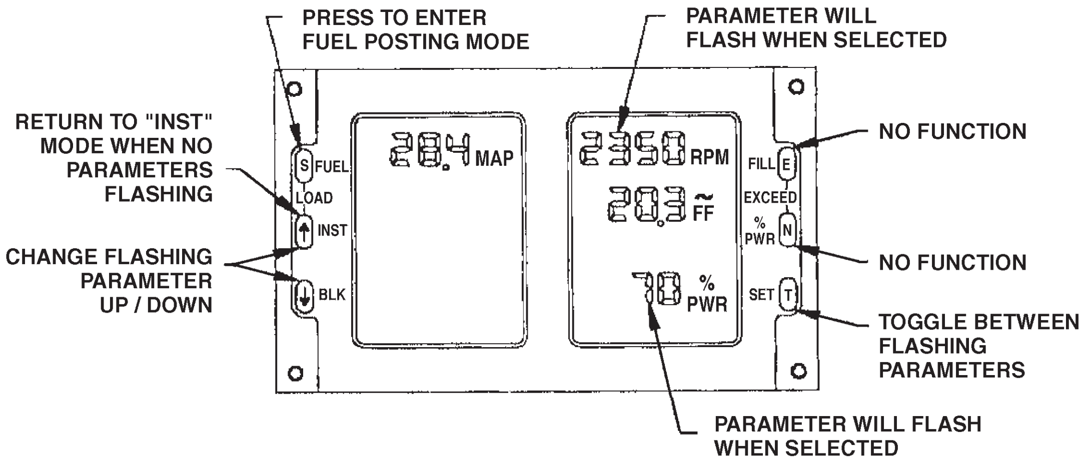


EXAMPLE OF ANALOG INDICATOR EXCEEDANCE REVIEW MESSAGE

THE ABOVE EXAMPLE INDICATES:
1) THE PRESENT INDICATOR IN REVIEW IS THE "OP" INDICATOR
2) THE PEAK EXCEEDANCE EVENT POSTED, " 118 PSI ", WAS THE LAST RECORDED
3) THERE ARE 5 EXCEEDANCE EVENTS IN MEMORY FOR THIS INDICATOR

NOTE: THE ANALOG PRESENTATION IS FOR REFERENCE ONLY.
DIAL CONFIGURATION IS SUBJECT TO CHANGE.

Exceedance Review - Analog Indicator
Figure 12



Percent Power Mode
Figure 13

Effectivity
4636132 thru 4636374

(15) Percent Power

EDI only - S/N's 4636132 thru 4636374

(a) Another feature of the EDI is the percent power mode ("Figure 2" B). To enter this mode momentarily depress the percent power (% PWR) button (middle button on the right side of the EDI). This will bring up the percent power page with the following parameters listed. Manifold Pressure (MAP), RPM, Approximate Fuel Flow (~FF) and Percent Power (% PWR). A desired percent power setting can be chosen by first depressing the set button on the lower right side of the EDI. Depressing this button toggles between the percent power and RPM parameters, causing the active parameter to flash. Once the percent power parameter is active the power can be cycled up or down by depressing the appropriate up and down arrow keys on the left side of the display. The % power range is that which can be selected is from 45% to 80% in 5% increments.

(b) As % power is changed, a suggested approximate value for manifold pressure and fuel flow will be displayed. If a different RPM is desired, the select button can be depressed to toggle the active parameter to the RPM display, and the up/down arrow buttons can be used to vary the RPM in 100 RPM increments. This variation in RPM changes expected values of MAP and Fuel Flow accordingly. Once the desired % PWR and RPM combination are chosen pressing of the instrument button (INST) will return to the instrument display mode.

C. Calibration

Transicoil Electronic Module Instrument System (EMIS) components (i.e. - QDI/EDI and Dual Analog Module Indicators) can be calibrated against the criteria in "Chart 3", except for Fuel Quantity and Cylinder Head Temperature (CHT).

Fuel Quantity calibration specifications and procedures are in Section 28-40-00.

CHT indicator calibration specifications and procedures are in Section 77-20-00.

CHART 3
Electronic Module Instrument System (EMIS) CALIBRATION

INDICATOR TYPE	SENSOR FULL SCALE OUTPUT (UNLESS NOTED)	CALIBRATION POINTS EACH INDICATOR SHALL MEET THE CRITERIA SPECIFIED BELOW WITH ITS APPLICABLE SENDER UNIT						
OAT	84.23 TO 121.32 OHMS (- 40 TO 55 °C)	TEST POINT	°C	- 40	-20	0	20	55
		MAXIMUM OHMS		85.07	93.00	100.84	108.63	122.16
		MINIMUM OHMS		83.47	91.32	99.16	106.95	120.48
MAP [1]	1.786 TO 4.536 VDC (10 TO 45 IN HG)	TEST POINT IN HG(ABS)		10	20	30	42	45
		MAXIMUM VDC		1.814	2.599	3.385	4.328	4.564
		MINIMUM VDC		1.759	2.544	3.330	4.273	4.509
RPM [2]	12.50 HZ TO 67.50 HZ (500 TO 2700 RPM)	TEST POINT	RPM	500	1400	2000	2500	2700
		MAXIMUM HZ		13.05	35.55	50.55	63.05	68.05
		MINIMUM HZ		11.95	34.45	49.45	61.95	66.95
TIT	0 TO 1850 °F - 0.692 mV TO 41.657 mV	TEST POINT	°F	200	1200	1600	1750	1800
		MAXIMUM °F		218.5	1218.5	1618.5	1768.5	1818.5
		MINIMUM °F		181.5	1181.5	1581.5	1731.5	1781.5
FUEL FLOW	53.831 TO 302.8 HZ (8 TO 45 GPH)	TEST POINT	GPH	8	15	25	35	45
		MAXIMUM HZ		56.321	103.423	170.712	238.001	305.290
		MINIMUM HZ		51.341	98.443	165.732	233.021	300.310
CHT [3]	60.09 TO 143.80 OHMS (NON-LINEAR)	SEE SECTION 77-20-00						
CHT [4]	417.87 mV TO 1.00 V (NON-LINEAR)	SEE SECTION 77-20-00						
OIL TEMPERATURE	84.50 TO 143.45 OHMS (NON-LINEAR)	TEST POINT	°F	0	140	180	245	270
		MAXIMUM OHMS		85.00	112.88	121.92	137.71	144.15
		MINIMUM OHMS		84.01	111.70	120.66	136.35	142.75
OIL PRESSURE	1.0 TO 4.2 VDC (0 TO 120 PSI)	TEST POINT	PSI	0	25	70	115	120
		MAXIMUM VDC		1.032	1.699	2.899	4.099	4.232
		MINIMUM VDC		0.968	1.635	2.835	4.035	4.168
VAC [5, 6]	3.0 TO 5.0 VDC (3.0 TO 6.0 IN HG)	TEST POINT	IN HG	3.0	4.8	5.0	5.2	6.0
		MAXIMUM VDC		3.020	4.220	4.353	4.487	5.020
		MINIMUM VDC		2.980	4.180	4.313	4.447	4.980
VAC [5, 7]		TEST POINT	IN HG	3.0	4.5	5.0	5.2	6.0
		MAXIMUM VDC		3.020	4.020	4.353	4.487	5.020
		MINIMUM VDC		2.980	3.980	4.313	4.447	4.980
FUEL QUANTITY		SEE SECTION 28-40-00						
		(PIR-PS50183 S.)						

Calibrate with 28Vdc at room temperature.

1. When the MAP indicator receives a signal from 41.1 IN HG to 42.9 IN HG, the digital indicator will snap to 42 IN HG. The analog MAP indicator will continue to display the actual IN HG. When calibrating MAP indicator, shorting pins 23 and 25 on the QDI/EDI will result in a digital increment of 0.1 IN HG, allowing a more accurate set point.
2. When the RPM indicator receives a signal from 2460 RPM to 2540 RPM, the digital indicator will snap to 2500 RPM. The analog RPM indicator will continue to display the actual RPM.
3. P/N 599-588.
4. P/N's 602-229 or 602-247.
5. When the vacuum indicator receives a signal below 3.0 IN HG, the digital indicator will snap to 0 IN HG.
6. P/N's 599-588 or 602-229.
7. P/N 602-247.

2. TIT Indicator Check

In PA-46-350P S/N's 4636021 and up and all PA-46R-350T, electro-mechanical TIT indicators are no longer used. Calibration of the electronic indication systems (Transicoil, Avidyne, and/or Garmin G1000/ G1000 NXi) are not field adjustable and any adjustments must be performed by the authorized service facilities of the original system manufacturer.

To check the accuracy of the TIT indications on the aircraft, a handheld thermocouple calibrator (such as Omega's Models CL3515R or CL300A) can be used to provide an accurate simulated temperature output to the TIT indicator/processor that will verify that the system can correctly display the TIT probe output. The instructions that are provided with the thermocouple calibrator should be used to provide the connection and appropriate signal to the TIT indicator.

NOTE: Perform this test with the two wire TIT harness removed from the TIT probe.

NOTE: The temperature calibrator is to be connected to the two wire harness that routes to the TIT indicator/indicator processor.

NOTE: Maintain polarity between the calibrator and DAU.

3. Avidyne Entegra IFDS

In PA-46-350P S/N's 4636375 thru 4636459, 4636461 thru 4636462, and 4636481; and, PA-46R-350T S/N's 4692001 thru 4692133, 4692141, 4692149 and 4692153: engine instrumentation is integrated into the Avidyne Entegra system. See 28-40-00, 34-22-00, and 91-77-40 for more information.

4. Garmin G1000 /G1000 NXi Engine Indication System (EIS)

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

In PA-46-350P S/N's 4636460, 4636463 and up; and PA-46R-350T S/N's 4692134 and up: engine instrumentation is integrated into the Garmin G1000/G1000 NXi Integrated Avionics System.

The G1000/G1000 NXi Engine Indication System (EIS) displays electrical, fuel, engine, pressurization (if installed), and flap position information on the left side of the Multi Function Display (MFD).

EIS information is presented using gauges, horizontal and vertical bar indicators, slide bars, and digital readouts. Green ranges on the instrument scales indicate normal ranges of operation; yellow and red bands indicate caution and warning, respectively. During normal operating conditions, an instrument's title appears in white and the readout text is green. When an unsafe operating condition occurs, the title and readout color change to yellow or flash red, indicating a caution or warning. If the sensor data for a parameter becomes invalid or unavailable, a red "X" or yellow "X" is displayed across the indicator and/or readout.

The EIS displays manifold pressure, propeller speed, and turbine inlet temperature, using arc gauges.

Engine data is collected by the Engine/Airframe Unit (GEA) and is displayed on the Multi-Function Display (MFD) and Primary Flight Display (PFD). The systems are addressed in 34-25-01 (G1000) or 34-25-02 (G1000 NXi.

The Engine/Airframe Unit (GEA) provides engine/airframe data to the G1000/G1000 NXi system. Data received from transducers/sensors is processed and sent to GIA (via RS-485 digital interface), and subsequently to the MFD. Engine parameters are normally displayed on the MFD. In the event of a MFD failure, all engine instruments can be displayed on the PFD's. The GEA is located behind the instrument panel and is mounted in a vertical orientation. Power is received from the main bus or the Emergency bus.

For more information see 34-25-01 (G1000) or 34-25-02 (G1000 NXi).

CHAPTER

79

OIL

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CHAPTER 79

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
79-List of Effective Pages	1	Nov 30/17			
	2	Sep 15/09			
79-Table of Contents	1	Nov 30/17			
	2	Sep 15/09			
79-20-00	1	Sep 15/09			
	2	Sep 15/09			
79-30-00	1	Nov 30/17			
	2	Nov 30/17			
	3	Feb 15/14			
	4	Sep 15/09			

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CHAPTER 79 - OIL

TABLE OF CONTENTS

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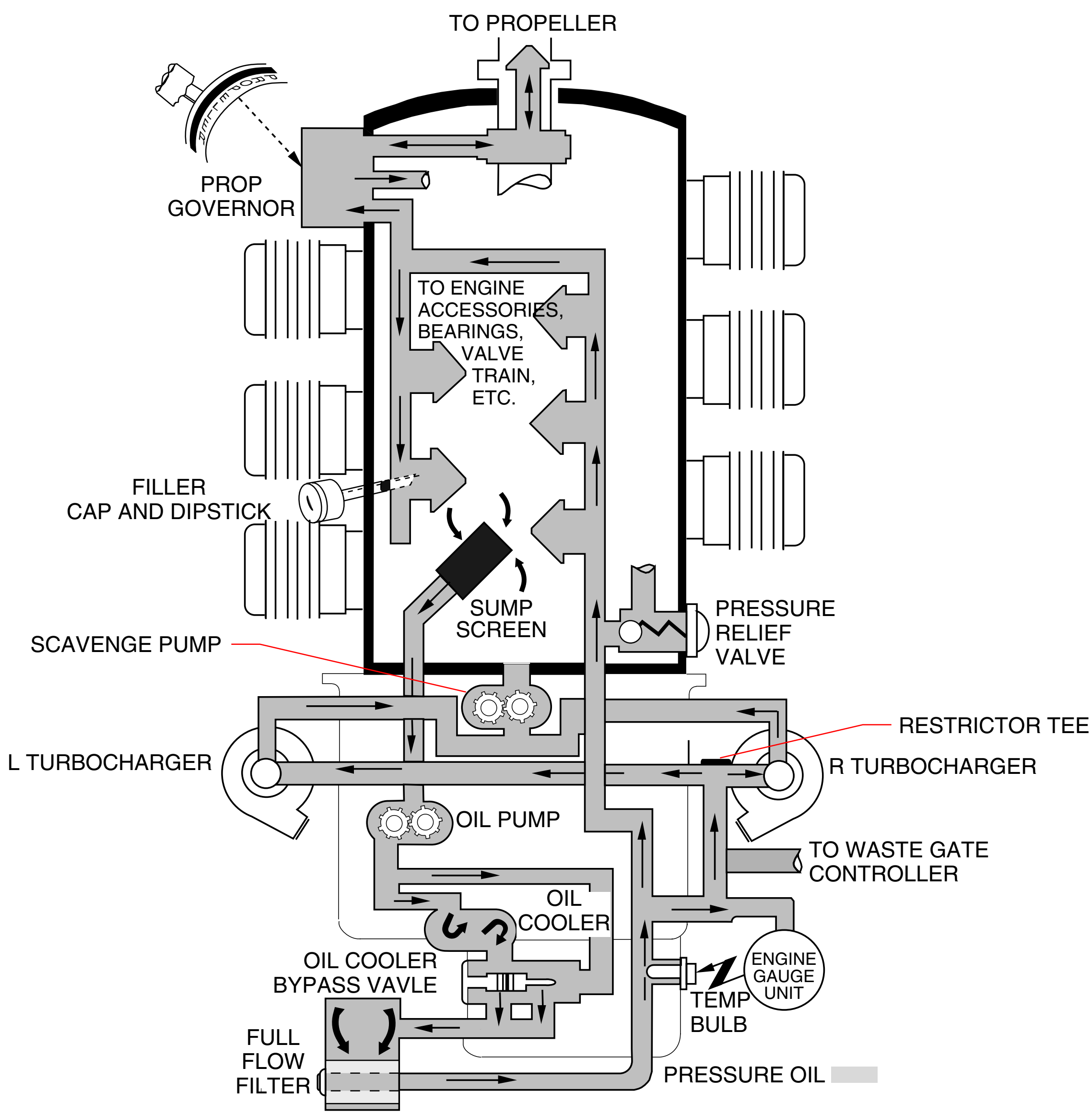
DISTRIBUTION

Oil System

Oil is drawn from the sump through a suction tube to the intake side of the engine-driven, gear-type, oil pump. Outlet oil is directed to a full-flow, replaceable-element oil filter. A bypass valve is incorporated in the filter. The oil pump has an oil pressure-relief valve in the housing. A second gear-driven pump, located below the starter, scavenges oil from the turbochargers.

Engine oil is cooled by ram air passing through the oil cooler on the rear of the engine. Oil is distributed throughout the engine, providing lubrication, cooling and oil supply to the propeller governor. Oil temperature and pressure information is available from the combination gauge on the bottom right of the pilot's instrument panel. Engine crankcase gases are discharged to an air/oil separator behind the oil cooler and are then vented out the left exhaust stack.



Engine Oil System Fluid Diagram
Figure 1

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INDICATING

NOTE: PA-46-350P S/N's 4636021–4636374 incorporate the Transicoil Electronic Module Instrument System (EMIS). See 77-40-00 for more information.

NOTE: In PA-46-350P S/N's 4636375 and up; and PA-46R-350T S/N's 4692001 and up: engine data is collected by the Data Acquisition Unit (DAU) (Avidyne) or Engine/Airframe Unit (GEA) (Garmin G1000/G1000 NXi) and is normally displayed on the Multi-Function Display (MFD). In the event of a MFD failure, all engine instruments can be displayed on the Primary Flight Displays (PFD's). See 34-22-00, 34-25-01 (G1000), or 34-25-02 (G1000 NXi); 77-40-00, and 91-77-40 for more information.

1. Engine Oil Pressure

A. Gauge

(1) Description

The oil pressure gauge is mounted next to the landing gear lever on the instrument panel. This gauge will indicate the amount of oil pressure available at the pressurized engine oil passage.

(2) Troubleshooting

See “Chart 1”.

(3) Removal and Installation

See Face-mounted Instruments, 39-10-00.

CHART 1
TROUBLESHOOTING ENGINE OIL PRESSURE GAUGE

Trouble	Cause	Remedy
Excessive error at zero.	Pointer loose on shaft.	Replace instrument.
Excessive scale error.	Improper calibration adjustment.	Replace instrument.
Excessive pointer oscillation.	Air in line or rough engine relief.	Disconnect line and fill with light oil. Check for leaks. If trouble persists, clean and adjust relief valve.
Sluggish operation of pointer or pressure fails to build up.	Engine relief valve open.	Clean and check.

B. Transducer

S/N's 4636021 and up; 4692001 and up.

The oil pressure transducer is mounted on an engine mount tube on the right side of the engine compartment. (See "Figure 1".)

(1) Removal

(a) Disconnect the electrical leads.

(b) Unscrew the transducer from the T-fitting on the engine mount tube.

(c) Catch spillage and cover hole to prevent foreign matter from entering oil line.

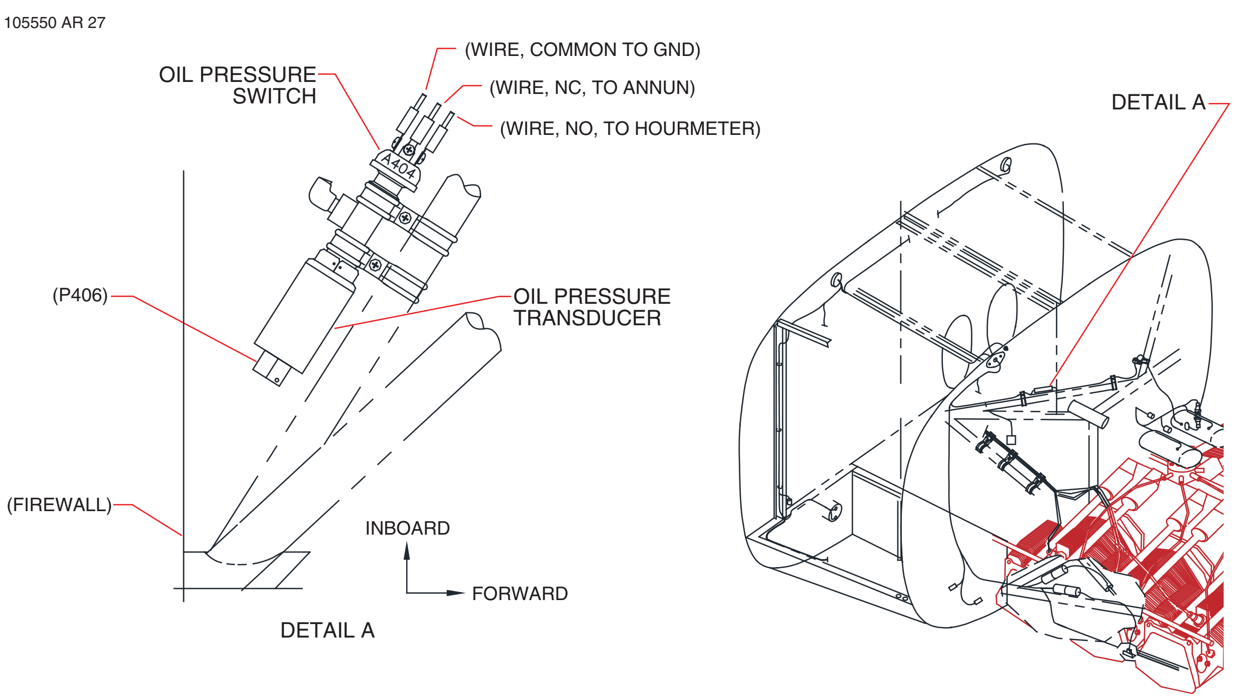
(2) Installation

(a) Screw the transducer into the T-fitting. Torque is to be 75 in·lbs max.

(b) Perform resistivity check between sender's body and aircraft ground, resistance should be less than .003 ohms.

(c) Reconnect the electrical leads.

(d) Perform operational check.



Engine Oil Pressure Transducer
Figure 1

2. Engine Oil Temperature

Gauge

(1) Description

The oil temperature indicator is mounted next to the landing gear lever on the instrument panel. This instrument will provide a temperature indication of the engine oil in degrees Fahrenheit. The instrument has a temperature bulb (heat sensitive probe) located between the magnetos.

(2) Troubleshooting

See “Chart 2”.

(3) Removal and Installation

See Face-mounted Instruments, 39-10-00.

CHART 2
TROUBLESHOOTING ENGINE OIL TEMPERATURE GAUGE

Trouble	Cause	Remedy
Instrument fails to show any reading.	Broken or damaged bulb. Wiring open.	Check engine unit and wiring to instrument.
Excessive scale error.	Improper calibration adjustment.	Repair or replace.
Pointer fails to move as engine Is warmed up.	Broken or damaged bulb or open wiring.	Check engine unit and wiring.
Dull or discolored marking.	Age	Replace instrument.

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CHAPTER

80

STARTING

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CHAPTER 80

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
80-List of Effective Pages	1	Mar 25/21			
	2	Sep 15/09			
80-Table of Contents	1	Mar 25/21			
	2	Sep 15/09			
80-10-00	1	Mar 25/21			
	2	Feb 15/14			
	3	Feb 15/14			
	4	Feb 15/14			
	5	Sep 15/09			
	6	Sep 15/09			
	7	Sep 15/09			
	8	Sep 15/09			
	9	Sep 15/09			
	10	Sep 15/09			
	11	Sep 15/09			
	12	Sep 15/09			

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CHAPTER 80 - STARTING

TABLE OF CONTENTS

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CRANKING

1. Cranking Limitations

Do not crank for more than 10 seconds. Allow 20 seconds for cool down between cranking attempts. Repeat no more than six (6) times. If start is not achieved on the sixth attempt, let starter cool for 30 minutes before reattempt.

2. Hartzell Engine Technologies (HET) Starter

Formerly Kelly Aerospace, Electrosystems, Prestolite.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

S/N's 4636001 thru 4636402.

A. Description

When the starting circuit is energized, battery current is applied to the starting motor terminal. Current flows through the field coils creating a strong magnetic field. At the same time, current also flows through the brushes and commutator, and finally through the armature to ground. The result is a high cranking torque, for a limited period of time, with a high current flow at a minimum loss of voltage.

The drive end gear of the armature mates with the reduction gear which drives the Bendix shaft. The Bendix drive is held in position on the shaft by a spiral pin. As the armature turns the reduction gear, the Bendix drive pinon meshes with the starter drive gear by inertia and action of the spiral grooves within the Bendix unit. A detent pin engages in a notch in the screw threads which prevents demeshing if the engine fails to start. When the engine does start and reaches a predetermined speed, centrifugal action forces the detent pin to release and allows the pinion to demesh from the starter drive gear.

B. Troubleshooting

See "Chart 1".

NOTE: In "Chart 1", for any remedy which requires disassembly and cleaning or repair of the starter; replacement with a new, or known good, unit is always an alternative solution.

NOTE: NEVER USE JUMPER CABLES to test voltage to the starter. The "toothed" jaws of jumper cables are meant to "bite" into soft, leaded terminals on car batteries, and simply WILL NOT provide enough contact with the starter terminal to supply the needed amperage to engage the starter properly.

NOTE: Use an analog voltage meter if you can. Digital meters take intermittent 'snap shots' of voltage. In situations where voltage is being supplied intermittently (even in rapid cycles), the digital meter will simply not provide the correct 'picture' of the aircraft's voltage situation.

C. Removal

CAUTION: TO PREVENT SHORT CIRCUITING, DISCONNECT THE GROUND CABLE FROM THE BATTERY BEFORE REMOVING THE STARTER FROM THE ENGINE.

(1) Disconnect ground cable from battery.

CAUTION: ROTATION OF THE STARTER POST WILL CAUSE INTERNAL DAMAGE TO THE STARTER. ALWAYS HOLD THE STARTER POST'S BOTTOM NUT IN PLACE WHEN TORQUING OR REMOVING THE TOP NUT.

(2) Disconnect the starter cable from the terminal post (or stud): Hold the bottom nut on the terminal (or starter) stud in place with a wrench to prevent the stud from rotating. Loosen the top nut.

(3) Remove one (1) each mounting bolt, internal tooth “star” lock washer, and flat washer; and three (3) each mounting nuts, internal tooth “star” lock washers, and flat washers.

(4) Lift off starter motor.

D. Installation

NOTE: In S/N’s 4636001 thru 4636402, if installing a Sky-Tec 31B23592 Starter, order Piper Service Kit No. 88405-002.

NOTE: In S/N’s 4636001 thru 4636326, if installing a Lycoming 31B22474 Starter, order Piper Kit No. 767-337.

(1) Clean all traces of rust, corrosion, or dirt from all mounting surfaces and mounting hardware. All ground points or straps must be clean and tight.

(2) Place starting motor in position with no stresses or binding forces being present and install three (3) each flat washers, internal tooth “star” lock washers, and mounting nuts; and one (1)each flat washer, internal tooth “star” lock washer, and mounting bolt. Torque bolt and nuts to 204 in.-lbs. or as specified in Lycoming SSP-1776.

CAUTION: ROTATION OF THE STARTER POST WILL CAUSE INTERNAL DAMAGE TO THE STARTER. ALWAYS HOLD THE STARTER POST’S BOTTOM NUT IN PLACE WHEN TORQUING OR REMOVING THE TOP NUT.

(3) Reinstall starter cable to starter terminal post (or stud): Hold the bottom nut on the terminal (or starter) stud in place with a wrench to prevent the stud from rotating. Tighten the top nut. Torque to 40 in.-lbs. or as specified in Lycoming SSP-1776.

(4) Reconnect ground cable to negative post of battery.

(5) Perform cranking tests, below.

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING - HET STARTER

Trouble	Cause	Remedy
Starter fails to operate.	Low battery charge.	Check and recharge if necessary.
	Defective or improper wiring or loose connections.	Refer to wiring diagram and check all wiring.
	Defective starter solenoid or control switch.	Replace faulty unit.
	Binding, worn, or improperly seated brush, or brushes with excessive side play.	Brushes should be a free fit in the brush boxes without excessive side play. Binding brushes and brush boxes should be wiped clean with a gasoline (undoped) moistened cloth. A new brush should be run in until at least 50% seated; however, if facilities are not available for running in brushes, then the brush should be properly seated by inserting a strip of No. 0000 sandpaper between the brush and commutator with the sanded side next to the brush.
CAUTION: DO NOT USE COARSE SANDPAPER OR EMERY CLOTH.		
		Pull sandpaper in the direction of rotation, being careful to keep it in the same contour as the commutator.
		After seating, clean thoroughly to remove all sand and metal particles to prevent excessive wear. Keep motor bearing free from sand or metal particles.
	Dirty commutator.	If commutator is rough or dirty, smooth and polish with No. 000 sandpaper. If too rough and pitted, remove and turn down. Blow out all particles.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING - HET STARTER

Trouble	Cause	Remedy
Starter fails to operate. (continued)	Shorted, grounded, or open armature.	Remove and replace with an armature known to be in good condition.
	Grounded or open field circuit.	Test and then replace with new part.
Starter operates at proper speed but fails to crank.	Faulty Bendix drive.	Remove Bendix drive assembly. Clean and check engine. Reinstall.
Low starter and cranking speed.	Worn, rough, or improperly lubricated motor or starter.	Disassemble, clean, inspect and relubricate, replacing ball bearings, if worn.
	See electrical causes listed under “Stater fails to operate,” above.	See remedies listed for “Stater fails to operate,” above.
Excessive arcing of starter brushes.	Binding, worn, or improperly seated brush or brushes, with excessive side play.	See information above dealing with this trouble.
	Dirty, rough, pitted or scored commutator.	Clean as outlined above.
	Grounded or open field circuit.	Test and replace defective parts.
Excessive wear and arcing of starter brushes.	Rough or scored commutator.	Remove and turn commutator down on a lathe.
	Armature assembly not concentric.	Reface commutator.

E. Cranking Tests

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

NOTE: NEVER USE JUMPER CABLES to test voltage to the starter. The "toothed" jaws of jumper cables are meant to "bite" into soft, leaded terminals on car batteries, and simply WILL NOT provide enough contact with the starter terminal to supply the needed amperage to engage the starter properly.

NOTE: Use an analog voltage meter if you can. Digital meters take intermittent 'snap shots' of voltage. In situations where voltage is being supplied intermittently (even in rapid cycles), the digital meter will simply not provide the correct 'picture' of the aircraft's voltage situation.

The starting circuit should be inspected at regular intervals. The frequency should be determined by the type of starting conditions and the amount of starter usage. In any case, it is recommended that the following tests be conducted each six months or every 100 hours time-in-service.

(1) Check the battery with a hydrometer to make sure it is fully charged and filled to the proper level. A load test should be made on the battery to verify proper condition before proceeding.

(2) Check all starter circuit wiring, making sure all connections, including battery terminals, are clean and tight and that all insulation is sound and complete.

(3) A voltage loss test should be made to locate any high-resistance connections that would impair starting motor electrical efficiency. Using a low-reading voltmeter scale while cranking engine (or at approximately 100 amperes current flow) measure for the following limits:

(a) Voltage loss from the insulated (positive) battery post to the starter motor terminal = 0.3 volt maximum.

(b) Voltage loss from the battery negative (ground) terminal to the starter motor frame = 0.1 volt maximum.

NOTE: If voltage loss exceeds the above limits, measure the voltage drops across all connections to discover the faulty connection. When within the maximum limits proceed to next step.

(4) The starter motor should be operated for several seconds with the ignition OFF. The starter motor engagement should be prompt and the motor should turn freely at a uniform speed without binding or producing unusual sounds.

(5) Re-engage the starter two or three times, listening for prompt engagement, without the clashing of gears, and to determine that the pinion disengages properly when the starter switch is released.

3. Sky-Tec Starter

See “Figure 1”.

Factory installed 2007 and up; and authorized as service replacments in S/N’s 4636001 and up.

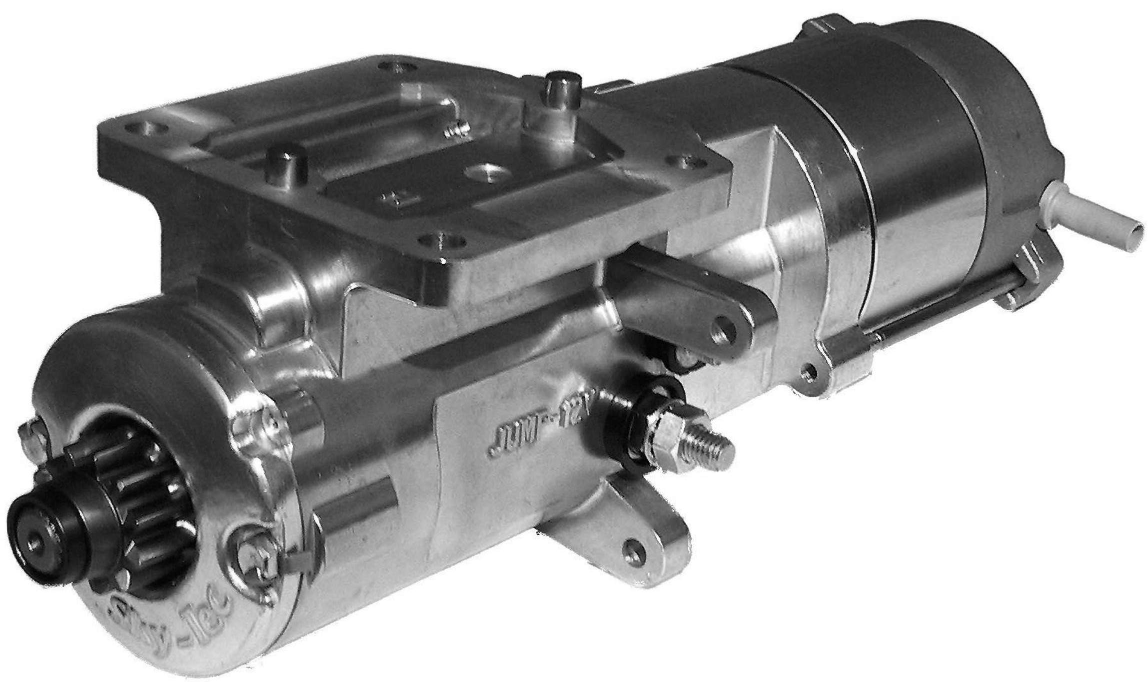
NOTE: In S/N’s 4636001 thru 4636402, if installing a Sky-Tec 31B23592 Starter, order Piper Service Kit No. 88405-002.

A. Description

The Sky-Tec 149-NL high-performance starters have significant differences from the earlier starters made by Kelly Aerospace, Electrosystems, and Prestolite. The Sky-Tec starters use a more modern, more reliable, bendix-free design. No periodic maintenance is required. The recommended TBO is 2700 hours. Engine cranking speed may be up to twice as fast as with previous starters. The light weight of these starters does, however, provide less mass to extract heat from starter components. Accordingly, strict adherance to cranking limitations, below, is essential to prevent overheating.

B. Troubleshooting

See “Chart 2”.



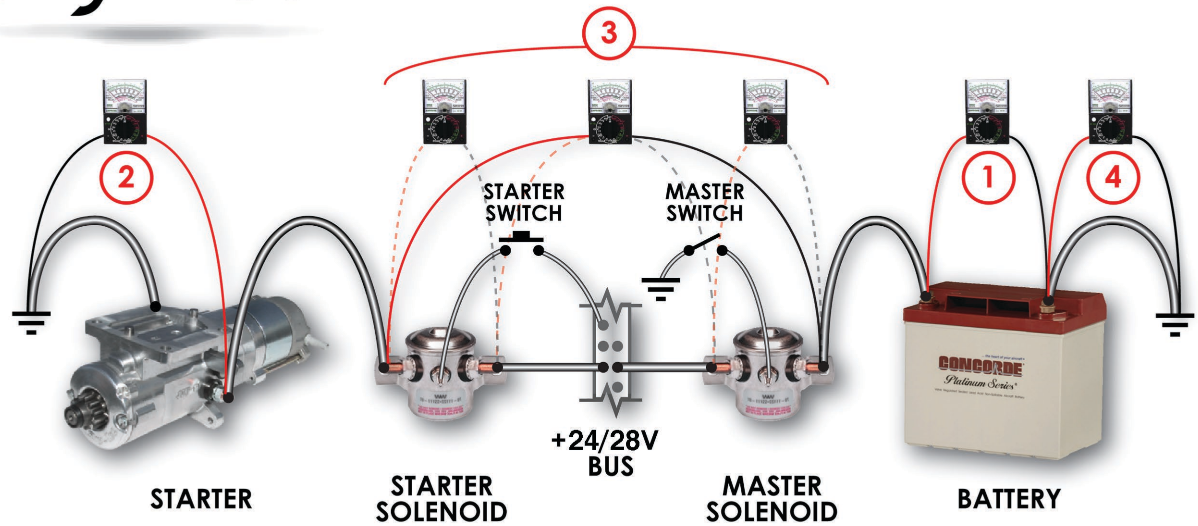
Sky-Tec High-Torque Inline Starter
Figure 1

CHART 2
TROUBLESHOOTING - SKY-TEC STARTER

Sky-Tec

AIRCRAFT STARTING SYSTEM TROUBLESHOOTING GUIDE

REV_3.0



IMPORTANT NOTES: DO NOT REMOVE THE STARTER FROM THE AIRCRAFT TO PERFORM TESTING! For accuracy and relevance, starter testing should be conducted in the aircraft while the starter is in its failure mode. Please do NOT use jumper cables to test starters - it will not provide useful information. Use an analog voltmeter if at all possible. An ohmmeter will not provide useful information - keep it simple & follow these easy steps:

Pretest Considerations: If possible, visually inspect the starter and/or interview pilot/operators for starting history. Indications of long cranking periods, burning odor or smoke from the starter, kickback(s), a cracked starter mount, a fast spinning starter w/no prop movement (**replace shear pin**), oil in starter, grinding noise or a damaged ring gear are generally indications that the starter is in need of repair and the following tests will not be helpful. Remove the starter for repair or overhaul (the shear pin is FIELD REPLACEABLE - do not return to Sky-Tec for shear pin repair).

Testing Relevance: The following testing procedure is most relevant to starters that are low performing including slow cranking and/or failure to crank the engine over a compression stroke. If a starter is damaged by overcranking or a stuck firewall solenoid, voltage in step one may read below acceptable levels thus incorrectly indicating a potential battery problem. Therefore, in such cases some consideration must be to the pretest conditions noted above (if it smells burned...). If the only effect of energizing the starter results only in an audible "click" with no prop movement, confirm step 2 to isolate problem to starter or starter contactor.

	TEST	RECORD	RESULT	ANALYSIS
STEP 1	Record voltage at the battery while cranking the starter in its failure mode.	____ VOLTS	Below 20.0 volts? Above 20.0 volts? - Proceed to Step 2	Questionable battery. See Testing Relevance above as a shorted (cooked) starter will pull voltage down appreciably. However, if starter rotates at all, this is generally not a shorted starter condition.
STEP 2	Record voltage at the starter while cranking the starter in its failure mode.	____ VOLTS	No voltage recorded	Starter OK. Test Starter Solenoid or switch (Step 3)
			Above 18.0 volts?	Suspect Starter.
			Below 18.0 volts and: - Difference between Step 1 and Step 2 exceeds 4.0 volts	Suspect Cables, terminals and/or solenoids - proceed to Steps 3 & 4
			- Difference between Step 1 and Step 2 less than 4.0 volts	Borderline Condition - Call Sky-Tec with test results to discuss. If happens more when cold, suspect battery. When hot, suspect cables/terms/sols.

OPTIONAL:

STEP 3	Record voltages between each and every cable terminal and across solenoids while cranking the starter in its failure mode.	Assuming the voltage difference noted in Step 2 exceeded 4.0 volts, flush out any appreciable loss in voltage in any cable or solenoid by placing the meter along each link in the diagram. A tight electrical system will lose no more than 0.5 volts between the battery and starter. Be sure to conduct these tests while cranking the starter in its failure mode.
STEP 4	Record voltages between battery & ground and starter & ground while cranking the starter in its failure mode.	If no appreciable loss of voltage is noted in Step 3, flush out the integrity of all electrical system grounds. Pay close attention to battery and engine grounds. Clean up or repair any questionable ground connections and re-test.

C. Installation as a Service Replacement

NOTE: In S/N's 4636001 thru 4636402, if installing a Sky-Tec 31B23592 Starter, order Piper Service Kit No. 88405-002.

Engines with high-time magnetos and weak electrical systems should be thoroughly inspected prior to installing any lightweight, high-torque starter. Failure to do so may make your aircraft susceptible to poor starter performance or damage from kickbacks.

(1) Ensure the jumper between the small "S" terminal and the larger power terminal is removed as shown in "Figure 2". Cover each terminal with a MS2571-1S or MS2571-2S (or equivilent) terminal nipple, as appropriate.

(2) Inspect the Magneto system thoroughly.

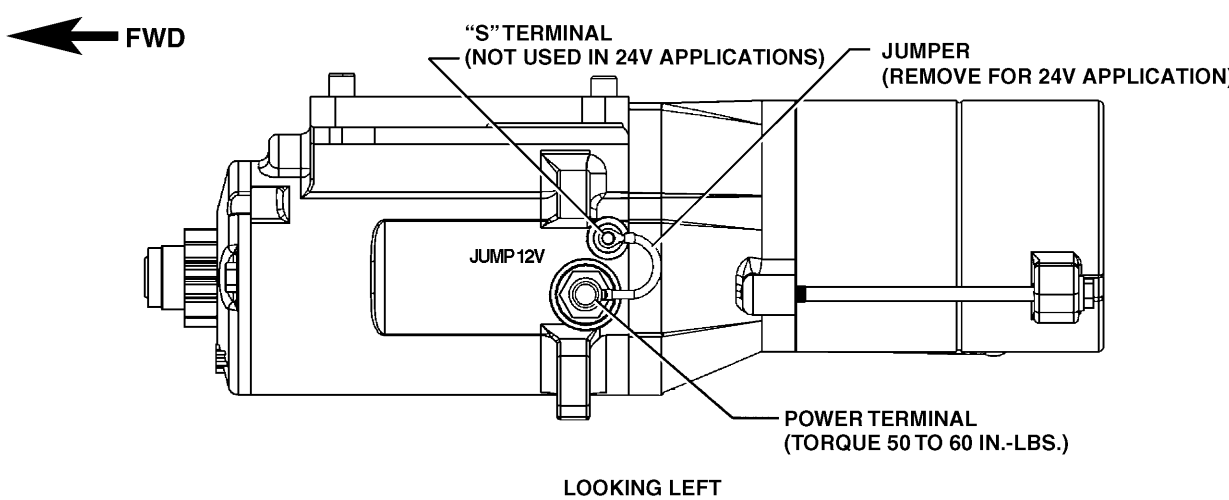
Make sure that the magnetos are in compliance with all service bulletins. Make sure that the magneto impulse coupling is within service specifications. Kickbacks will shear a drive pin. Check the impulse couplers prior to attempting even a single start.

(3) Inspect the voltage supply system.

If the Sky-Tec starter fails to turn the engine significantly faster than the OEM starter, immediately inspect the aircraft's voltage supply system. Lightweight starters can require as much as 45% more output from the battery – or as many as 300 Amps during initial cranking.

(4) Troubleshoot the entire voltage system

All Sky-Tec starters are tested to verify power output before leaving the factory. If a new starter does not spin quickly or seems weak, see "Chart 2". Do not return a "weak" starter without first completing the troubleshooting procedure in "Chart 2".



Jumper Removal for 24 Volt Applications
Figure 2

D. Removal

CAUTION: TO PREVENT SHORT CIRCUITING, DISCONNECT THE GROUND CABLE FROM THE BATTERY BEFORE REMOVING THE STARTER FROM THE ENGINE.

(1) Disconnect starter cable from power terminal post.

(2) Remove one (1) each mounting bolt, internal tooth “star” lock washer, and flat washer; and three (3) each mounting nuts, internal tooth “star” lock washers, and flat washers.

(3) Lift off starter motor.

E. Installation

(1) Clean all traces of rust, corrosion, or dirt from all mounting surfaces and mounting hardware. All ground points or straps must be clean and tight.

(2) Place starting motor in position with no stresses or binding forces being present and install three (3) each flat washers, internal tooth “star” lock washers, and mounting nuts; and one (1) each flat washer, internal tooth “star” lock washer, and mounting bolt. Torque bolt and nuts to 100 in.-lbs.

CAUTION: TAKE CARE NOT TO OVER-TORQUE THE POWER TERMINAL POST NUT. THE POWER TERMINAL POST IS COPPER AND CAN EASILY BE STRIPPED.

(3) Reinstall starter cable to power terminal post using an internal tooth “star” lock washer or split lock washer, as desired. Torque the power terminal post nut as specified in “Figure 2” and reinstall terminal nipple.

(4) Reconnect ground cable to negative post of battery.

F. Shear Pin Replacement

See “Figure 3”.

The shear pin is designed as the weak link to minimize starter and ring gear damage in the event of a premature ignition misfire event (i.e. - kickback). If the starter pinion engages the flywheel and the starter motor spins (or sounds like it’s spinning) and the starter’s pinion gear does not rotate (and neither does the flywheel, engine, and propeller) then the shear pin has done its job and must be replaced.

(1) Remove starter from airplane per Removal, above.

(2) Lift the insulating boot on the motor lead and remove the M8 locking nut with a 12mm wrench and remove the motor lead wire from the stud.

(3) Remove the two M5 x 25MM hex screws and locking tab washers holding the motor adapter to the main mount.

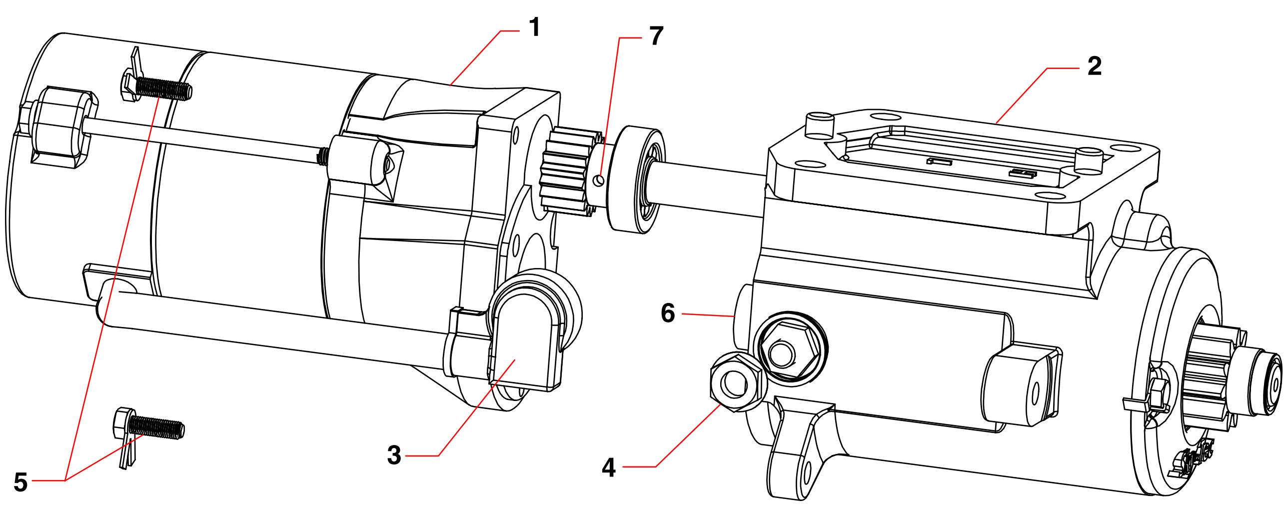
(4) Separate the two assemblies being careful not to dislodge the solenoid plunger residing in the main mount.

(5) Replace the shear pin in the lay-shaft.

(6) Reassemble the motor adapter and main mount, installing the two M5 screws and torquing them as specified in “Figure 3”.

(7) Reconnect the motor lead, secure with the M8 locking nut, and cover with the insulating boot.

(8) Install starter in airplane per Installation, above.



1. **MOTOR ADAPTER**
2. **MAIN MOUNT**
3. **INSULATING BOOT**
4. **NUT (M8 LOCKING)**
5. **HEX SCREWS (M5 X 25MM) AND LOCKING TAB WASHERS (TORQUE TO 50 IN.-LBS.)**
6. **SOLENOID PLUNGER**
7. **LAY-SHAFT (SHEAR PIN GOES HERE)**

Shear Pin Replacement
Figure 3

4. Starter Control Circuit

   A. Inspect the control circuit wiring between the battery, solenoid and manual starting switches for breaks, poor connections and faulty insulation. Tighten all connections and make sure solenoid is firmly mounted and makes a good ground connection.

   B. Check the voltage loss across the switch contacts during normal starting. If loss is in excess of 0.2 volts per 100 amperes, the solenoid should be replaced.

   C. If solenoid fails to operate when the manual starting switch is turned on or if it fails to release when the manual starting switch is released, it should be removed and tested to specifications, if available. If either opening or closing voltages are not to specifications, or tests cannot be performed, replace the solenoid.

5. Starting with External Power

   See 24-40-00.

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CHAPTER

81

TURBINES

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CHAPTER 81

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
81-List of Effective Pages	1	Mar 25/21			
	2	Sep 15/09			
81-Table of Contents	1	Mar 25/21			
	2	Sep 15/09			
81-20-00	1	Feb 15/14			
	2	Sep 15/09			
	3	Sep 15/09			
	4	Feb 15/14			
	5	Feb 15/14			
	6	Mar 25/21			
	7	Mar 25/21			
	8	Mar 25/21			

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CHAPTER 81 - TURBINES

TABLE OF CONTENTS

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TURBO-SUPERCHARGER

1. Turbocharger System

   A. Description

   This chapter contains information on the removal, installation and adjustment of the various components of the turbocharging and exhaust system.

   The Lycoming TIO-540-AE2A engine is equipped with dual turbochargers, one on each side of the engine. Each turbocharger extracts exhaust energy from its respective bank of cylinders to pressurize the induction air.

   Air is taken from the induction inlet louvers into the induction air box. Here it is filtered and divided for distribution to the left and right turbocharger compressors. The compressed air then flows through an aftercooler where its air temperature is reduced. Downstream from the aftercoolers, theairflows join at the Y junction of the intake tubes, then pass through the throttle butterfly valve, and are divided to individual intake pipes to each cylinder.

   Metered fuel is injected into the cylinder head upstream of the intake valve. After combustion in the cylinder, exhaust gases flow into the exhaust manifold and to the turbocharger turbines, where exhaust energy is extracted to drive the turbines which, in turn, drive or turn the compressor.

   B. Troubleshooting

   See “Chart 1”.

   C. Wastegate Controller Installation

   WARNING: WHEN INSTALLING THE WASTEGATE CONTROLLER, ENSURE THE FUEL INJECTOR LINKAGE ROD MOVES FREELY AND CLEARLY THROUGHOUT ITS FULL RANGE OF MOTION. SPECIFICALLY LOOK FOR INTERFERENCE BETWEEN THE WASTEGATE CONTROLLER OIL LINES AND FITTINGS AND THE FUEL INJECTOR LINKAGE ROD. MIS-CLOCKING OF THE OIL LINE FITTINGS CAN CAUSE INTERFERENCE.

   D. Variable Pressure Controller Installation

   WARNING: WHEN INSTALLING THE VARIABLE PRESSURE CONTROLLER, ENSURE THE FUEL INJECTOR LINKAGE ROD MOVES FREELY AND CLEARLY THROUGHOUT ITS FULL RANGE OF MOTION. SPECIFICALLY LOOK FOR INTERFERENCE BETWEEN THE VARIABLE PRESSURE CONTROLLER OIL LINES AND FITTINGS AND THE FUEL INJECTOR LINKAGE ROD.

   LIKEWISE, ENSURE THE LINKAGE ATOP THE VARIABLE PRESSURE CONTROLLER MOVES FREELY AND CLEARLY THROUGHOUT ITS FULL RANGE OF MOTION WITHOUT ANY CONTACT WITH THE LOWER VACUUM PUMP COOLING SHROUD / BLAST HOSE.

2. Turbocharger Pre - Lubrication

   Following engine oil and filter changes, and following any prolonged period of non-operation, pre-lubricate the turbochargers as follows:

   A. Temporarily disconnect the engine oil supply lines from the turbocharger oil inlet ports and the engine air duct from the compressor housing inlets.

   B. Fill the turbocharger oil inlet ports with clean engine oil and manually turn the compressor wheels several revolutions to coat all journal and bearing surfaces with oil. Reconnect the air ducts.

   C. Place suitable containers under the disconnected engine oil supply lines to avoid spilling oil on the engine.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING TURBOCHARGER SYSTEM

Trouble	Cause	Remedy
Excessive noise or vibration.	Improper bearing lubrication.	Supply required oil pressure. Clean or replace oil line; clean oil strainer. If trouble persists, overhaul turbocharger.
	Leak in engine intake or exhaust manifold.	Tighten loose connections or replace manifold gaskets as necessary.
	Dirty impeller blades.	Disassemble and clean.
Engine will not deliver rated power.	Clogged manifold system.	Clean all ducting.
	Foreign material lodged in compressor impeller or turbine.	Disassemble and clean.
	Excessive dirt build-up in compressor.	Thoroughly clean compressor assembly. Service air cleaner and check for leakage.
	Leak in engine intake or exhaust.	Tighten loose connections or replace manifold gaskets as necessary.
	Rotating assembly bearing seizure.	Overhaul turbocharger.
	Restriction in return lines from actuator to exhaust bypass controller.	Remove and clean lines.
	Exhaust bypass controller is in need of adjustment.	Have exhaust bypass controller adjusted.
	Oil pressure too low.	Tighten fittings. Replace lines, or hoses, increase oil pressure to desired pressure.
	Inlet orifice to actuator clogged.	Remove inlet line at actuator and clean orifice.
	Exhaust bypass controller malfunction.	Replace unit.
	Exhaust bypass butterfly not closing.	Low pressure. Clogged orifice in inlet to actuator.
		Butterfly shaft binding.
		Check bearings.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING TURBOCHARGER SYSTEM

Trouble	Cause	Remedy
Engine will not deliver rated power. (continued)	Turbocharger impeller binding, frozen or fouling housing.	Check bearings. Replace turbocharger.
	Piston seal in actuator leaking. Usually accompanied by oil leakage at drain line.	Remove and replace actuator or disassemble and replace packing.
Critical altitude lower than specified.	Controller not getting enough oil pressure to close the exhaust bypass.	Check pump outlet pressure, oil filters, external lines for leaks or obstructions.
	Chips under metering valve in controller holding it open.	Replace controller.
	Metering jet in actuator plugged.	Remove actuator and clean jet.
	Actuator piston seal failed and leaking excessively.	If there is oil leakage at actuator drain, clean cylinder and replace piston seal.
	Exhaust bypass valve sticking.	Clean and free action.
Engine surges or smokes.	Air in oil lines or actuator.	Bleed system.
	Controller metering valve stem seal leaking oil into manifold.	Replace controller.
	Clogged breather.	Check breather for restrictions to air flow.
NOTE: Smoke would be normal if engine has idled for a prolonged period.		
High deck pressure. (Compressor Discharge Pressure)	Controller metering valve not opening, aneroid bellows leaking.	Replace controller assembly or replace aneroid bellows.
	Exhaust bypass sticking closed.	Shut off valve in return line not working.
		Butterfly shaft binding. Check bearings.
		Replace exhaust bypass valve or correct linkage binding.
	Controller return line restricted.	Clean or replace line.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING TURBOCHARGER SYSTEM

Trouble	Cause	Remedy
High deck pressure. (Compressor Discharge Pressure)	Oil pressure too high.	Check pressure 75 to 85 psi (80 psi desired) at exhaust bypass actuator inlet. If pressure on outlet side of actuator is too high, have exhaust bypass controller adjusted.
	Exhaust bypass valve actuator piston locked in full closed position. (Usually accompanied by oil leakage at actuator drain line.)	Remove and disassemble actuator, check condition of piston and packing or replace actuator assembly.
NOTE: Exhaust bypass normally closed in idle and low power conditions. Should open when actuator inlet line is disconnected.		
	Exhaust bypass controller malfunction.	Replace controller.

D. Crank the engine, without allowing it to start.

E. When a steady flow of oil is obtained from the supply lines to show that air has been purged from the system, stop cranking. Reconnect the oil supply lines.

F. If the turbochargers are installed on a new or overhauled engine, with the initial run-in period not completed, proceed as follows:

CAUTION: BECAUSE CLEANING OF THE EXTRA SCREENS IS NOT PROVIDED FOR IN PERIODIC MAINTENANCE PROCEDURES FOR THE ENGINE, THE SCREENS MUST NOT BE LEFT IN PLACE AFTER THE INITIAL RUN-IN PERIOD. OPERATION OF THE TURBOCHARGERS WITH CLOGGED SCREENS WILL CAUSE TURBOCHARGER FAILURE DUE TO LACK OF LUBRICATION.

CAUTION: MAKE ENTRY IN AIRPLANE LOGBOOK TO ENSURE THAT SCREENS ARE REMOVED AFTER THE INITIAL ENGINE RUN-IN PERIOD HAS BEEN COMPLETED.

CAUTION: TAG LINES WITH A SUITABLE TAG TO REMIND MECHANIC THAT THE TEMPORARY SCREENS HAVE BEEN INSTALLED IN THE LINES.

(1) TEMPORARILY install a screen of 100 mesh or finer in the oil supply line to each turbocharger to trap metal particles from the engine during the initial run-in.

WARNING: OPERATION OF THE TURBOCHARGER WITHOUT ALL NORMALLY INSTALLED INLET DUCTS AND FILTERS CONNECTED WILL RESULT IN INJURY TO PERSONNEL AND DAMAGE TO EQUIPMENT FROM FOREIGN OBJECTS ENTERING THE TURBOCHARGER.

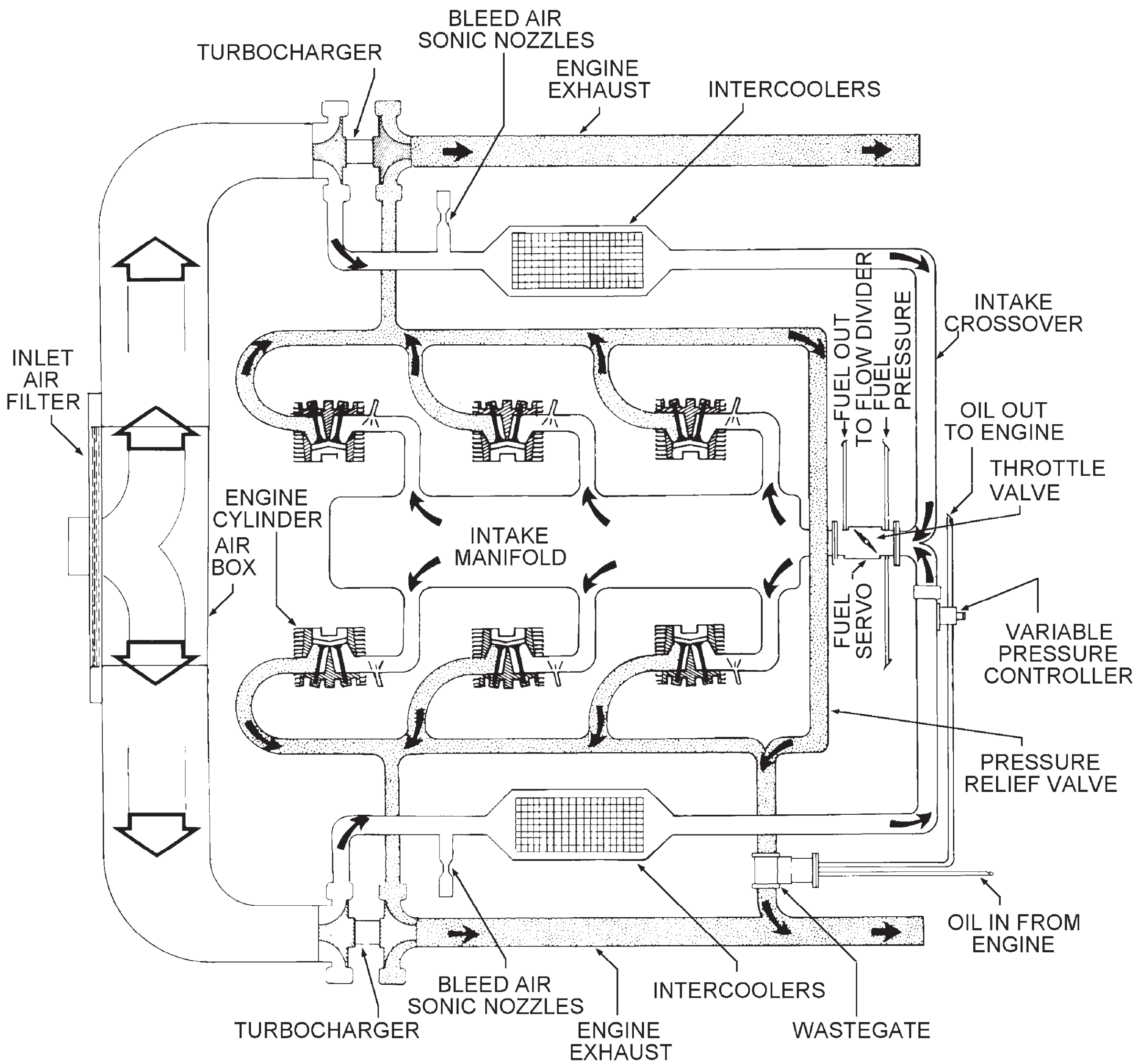
CAUTION: OPERATION OF THE ENGINE AT ANY SPEED FASTER THAN IDLE IMMEDIATELY AFTER START-UP CAN RESULT IN OIL LAG FAILURE OF TURBOCHARGER BEARINGS, ESPECIALLY IN COLD WEATHER OR AFTER A PROLONGED NON-OPERATIVE PERIOD.

(2) Remove the extra screens from the oil supply lines after the initial run - in period.

G. Verify that all air and exhaust inlet ducts and filters are properly installed.

H. Start engine and operate for 3 - 4 minutes at idle to allow engine oil pressure and flow to stabilize.

I. Check all oil connections for leakage, and tighten or repair as needed.



Turbocharger System Diagram
Figure 1

3. Inspections

A. Turbocharger and Exhaust System - Visual Inspection

(1) With the engine shut down, inspect all components of the air induction system for loose connections, cuts, cracks, punctures, corrosion or other evidence of deterioration. If evidence is found of any condition that could permit air leakage between the air cleaner and the turbocharger compressor, proceed as follows:

(a) Disconnect the air duct from the compressor air inlet and inspect the edges of the compressor wheel blades for damage. Also examine the wheel for excessive dirt buildup and oiliness which would indicate seal leakage, and check for evidence of wheel rub on the compressor housing.

(b) Replace damaged air ducting components.

(c) Reconnect the air ducting to the compressor air inlet and tighten attaching parts securely.

(2) Check the air duct connections between the compressor and the engine intake manifold and between the intake manifold and engine. Tighten any loose connections.

(3) Verify the engine air cleaner has been properly serviced (see Induction Air Filter, 12-10-00).

(4) Inspect the exhaust system for leakage, especially at the exhaust manifold connection to the turbocharger turbine inlet and at the engine exhaust manifold gasket.

(5) Disconnect the exhaust ducting from the turbine outlet and examine the turbine wheel blades for damage. Look for oil in the turbine wheel housing (indicative of seal leakage), and check for evidence of wheel rub on the housing.

(6) Check for oil leakage at the connections to the turbocharger oil inlet and drain ports. Tighten connections or replace gaskets, fittings, etc. as required.

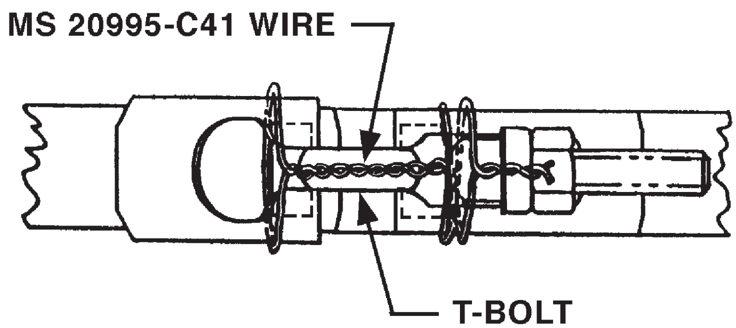
(7) On left-hand tail pipe, disconnect the engine crankcase breather tube and remove the lower tube assembly from the tail pipe at the slip joint. Remove any contaminate buildup accumulated. Accumulation may be very hard and may require scraping or sanding to remove. Ensure balance of breather is free of any restrictions to air flow.

(8) Remove the right-hand turbocharger exhaust heater muff, shroud hardware and shroud carefully as to not damage exterior insulation. Inspect for evidence of exhaust leakage. Perform leakage check in accordance with engine manufacture’s instructions using 5 psig clean pressurized air source with a soap and water solution. Replace exhaust heater muff assembly at any indication or doubt of leakage.

B. V-Band Coupling 100 Hour Inspection

See “Figure 2”.

Each 100 hours, inspect lockwiring on V-band couplings for condition and security. If lockwiring is found broken, inspect T-bolt for stretching, cracking, or any other damage. Replace coupling as required.



Lockwiring V-Band Couplings
Figure 2

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CHAPTER

91

CHARTS & WIRING DIAGRAMS

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CHAPTER 91

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
91-List of Effective Pages	1	Mar 25/21	91-X-REF	1	Nov 30/17
	2	Mar 25/21		2	Nov 30/17
	3	Mar 25/21		3	Nov 30/17
	4	Mar 25/21		4	Nov 30/17
	5	Mar 25/21		5	Nov 30/17
	6	Mar 25/21		6	Nov 30/17
	7	Mar 25/21		7	Nov 30/17
	8	Nov 30/17		8	Nov 30/17
				9	Nov 30/17
91-Table of Contents	1	Mar 25/21		10	Nov 30/17
	2	Mar 25/21		11	Nov 30/17
	3	Mar 25/21		12	Nov 30/17
	4	Mar 25/21		13	Nov 30/17
				14	Nov 30/17
91-10-00	1	Mar 25/21		15	Nov 30/17
	2	Mar 25/21		16	Nov 30/17
	3	Mar 25/21			
	4	Mar 25/21	91-21-20	1	Nov 30/17
	5	Mar 25/21		2	Nov 30/17
	6	Mar 25/21		3	Nov 30/17
	7	Mar 25/21		4	Nov 30/17
	8	Mar 25/21		5	Nov 30/17
	9	Mar 25/21		6	Nov 30/17
	10	Mar 25/21		7	Nov 30/17
	11	Mar 25/21		8	Nov 30/17
	12	Mar 25/21			
	13	Mar 25/21	91-21-30	1	Nov 30/17
	14	Mar 25/21		2	Nov 30/17
	15	Mar 25/21		3	Nov 30/17
	16	Mar 25/21		4	Nov 30/17
	17	Mar 25/21		5	Nov 30/17
	18	Mar 25/21		6	Nov 30/17
	19	Mar 25/21		7	Nov 30/17
	20	Mar 25/21		8	Nov 30/17
	21	Mar 25/21		9	Nov 30/17
	22	Mar 25/21		10	Nov 30/17
	23	Mar 25/21			
	24	Mar 25/21	91-21-40	1	Dec 15/14
	25	Mar 25/21		2	Dec 15/14
	26	Mar 25/21		3	Dec 15/14
	27	Mar 25/21		4	Dec 15/14
	28	Mar 25/21		5	Nov 30/17
				6	Dec 15/14

CHAPTER 91

LIST OF EFFECTIVE PAGES (continued)

CHAPTER 91

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
91-24-30 (cont.)	17	Dec 15/14	91-25-10 (cont.)	7	Dec 15/14
	18	Dec 15/14		8	Sep 15/09
	19	Nov 30/17		9	Dec 15/14
	20	Dec 15/14		10	Dec 15/14
	21	Nov 30/17		11	Dec 15/14
	22	Nov 30/17		12	Sep 15/09
	23	Nov 30/17			
	24	Nov 30/17	91-25-60	1	Feb 15/14
	25	Nov 30/17		2	Dec 15/14
	26	Nov 30/17		3	Dec 15/14
	27	Nov 30/17		4	Dec 15/14
	28	Nov 30/17		5	Dec 15/14
	29	Mar 25/21		6	Nov 30/17
	30	Mar 25/21		7	Nov 30/17
	31	Dec 15/14		8	Nov 30/17
	32	Dec 15/14			
	33	Dec 15/14	91-27-30	1	Mar 25/21
	34	Dec 15/14		2	Mar 25/21
	35	Dec 15/14		3	Mar 25/21
	36	Dec 15/14		4	Mar 25/21
	37	Dec 15/14		5	Mar 25/21
	38	Dec 15/14		6	Mar 25/21
	39	Dec 15/14		7	Mar 25/21
	40	Dec 15/14		8	Dec 15/14
	41	Dec 15/14			
	42	Nov 30/17	91-27-50	1	Nov 30/17
	43	Mar 25/21		2	Nov 30/17
	44	Mar 25/21		3	Nov 30/17
				4	Nov 30/17
91-24-60	1	Dec 15/14		5	Nov 30/17
	2	Dec 15/14		6	Nov 30/17
	3	Dec 15/14			
	4	Dec 15/14	91-27-60	1	Nov 30/17
	5	Nov 30/17		2	Nov 30/17
	6	Nov 30/17		3	Nov 30/17
	7	Nov 30/17		4	Nov 30/17
	8	Nov 30/17		5	Nov 30/17
	9	Nov 30/17		6	Nov 30/17
	10	Nov 30/17			
			91-28-20	1	Feb 21/20
91-25-10	1	Sep 15/09		2	Feb 21/20
	2	Dec 15/14		3	Feb 21/20
	3	Dec 15/14		4	Feb 21/20
	4	Dec 15/14		5	Feb 21/20
	5	Dec 15/14		6	Jul 1/10
	6	Dec 15/14			

CHAPTER 91

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
91-28-40	1	Dec 15/14	91-30-30 (cont.)	20	Mar 25/21
	2	Dec 15/14			
	3	Dec 15/14	91-30-40	1	Jul 1/10
	4	Dec 15/14		2	Dec 15/14
	5	Nov 30/17		3	Dec 15/14
	6	Dec 15/14		4	Dec 15/14
	7	Dec 15/14		5	Nov 30/17
	8	Dec 15/14		6	Jul 1/10
91-29-10	1	Dec 15/14	91-30-60	1	Dec 15/14
	2	Dec 15/14		2	Dec 15/14
	3	Dec 15/14		3	Dec 15/14
	4	Dec 15/14		4	Dec 15/14
	5	Nov 30/17		5	Dec 15/14
	6	Dec 15/14		6	Dec 15/14
	7	Dec 15/14		7	Nov 30/17
	8	Nov 30/17		8	Dec 15/14
91-30-10	1	Dec 15/14	91-31-20	1	Nov 30/17
	2	Dec 15/14		2	Dec 15/14
	3	Dec 15/14			
	4	Dec 15/14	91-31-30	1	Dec 15/14
	5	Dec 15/14		2	Dec 15/14
	6	Nov 30/17		3	Dec 15/14
	7	Dec 15/14		4	Dec 15/14
	8	Dec 15/14		5	Dec 15/14
				6	Nov 30/17
91-30-30	1	Nov 30/17			
	2	Nov 30/17	91-31-50	1	Dec 15/14
	3	Nov 30/17		2	Nov 30/17
	4	Nov 30/17		3	Nov 30/17
	5	Nov 30/17		4	Nov 30/17
	6	Nov 30/17		5	Nov 30/17
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	9	Feb 21/20		8	Nov 30/17
	10	Nov 30/17		9	Nov 30/17
	11	Mar 25/21		10	Nov 30/17
	12	Mar 25/21		11	Nov 30/17
	13	Mar 25/21		12	Dec 15/14
	14	Mar 25/21		13	Dec 15/14
	15	Mar 25/21		14	Dec 15/14
	16	Mar 25/21		15	Nov 30/17
	17	Mar 25/21		16	Dec 15/14
	18	Mar 25/21		17	Dec 15/14
	19	Mar 25/21		18	Dec 15/14

CHAPTER 91

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE
91-31-50 (cont.)	19	Dec 15/14
	20	Dec 15/14
	21	Dec 15/14
	22	Dec 15/14
	23	Dec 15/14
	24	Sep 15/09
	25	Dec 15/14
	26	Dec 15/14
	27	Dec 15/14
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	29	Dec 15/14
	30	Feb 15/14
91-32-60	1	Nov 30/17
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	3	Nov 30/17
	4	Nov 30/17
	5	Nov 30/17
	6	Nov 30/17
	7	Nov 30/17
	8	Nov 30/17
	9	Nov 30/17
	10	Nov 30/17
91-33-10	1	Jul 1/10
	2	Dec 15/14
	3	Dec 15/14
	4	Mar 25/21
	5	Sep 15/09
	6	Dec 15/14
	7	Dec 15/14
	8	Sep 15/09
	9	Dec 15/14
	10	Nov 30/17
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	18	Dec 15/14
	19	Dec 15/14
	20	Dec 15/14
	21	Dec 15/14
	22	Dec 15/14
	23	Nov 30/17
	24	Dec 15/14
91-33-10 (cont.)	25	Dec 15/14
	26	Dec 15/14
	27	Nov 30/17
	28	Dec 15/14
	29	Nov 30/17
	30	Nov 30/17
	31	Nov 30/17
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	10	Nov 30/17
	11	Nov 30/17
	12	Nov 30/17
91-33-40	1	Dec 15/14
	2	Dec 15/14
	3	Dec 15/14
	4	Dec 15/14
	5	Dec 15/14
	6	Dec 15/14
	7	Dec 15/14
	8	Nov 30/17
	9	Nov 30/17
	10	Dec 15/14
	11	Dec 15/14
	12	Dec 15/14
	13	Dec 15/14
	14	Dec 15/14
	15	Dec 15/14
	16	Dec 15/14
	17	Dec 15/14
	18	Nov 30/17

CHAPTER 91

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
91-33-40 (cont.)	19	Dec 15/14		29	Nov 30/17
	20	Dec 15/14		30	Nov 30/17
	21	Dec 15/14	91-34-20 (cont.)	31	Nov 30/17
	22	Dec 15/14		32	Nov 30/17
	23	Dec 15/14		33	Nov 30/17
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	4	Nov 30/17		41	Nov 30/17
	5	Nov 30/17		42	Nov 30/17
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	8	Dec 15/14		45	Nov 30/17
				46	Nov 30/17
91-34-20	1	Sep 15/09		47	Nov 30/17
	2	Dec 15/14		48	Nov 30/17
	3	Mar 25/21		49	Nov 30/17
	4	Dec 15/14		50	Nov 30/17
	5	Dec 15/14		51	Nov 30/17
	6	Nov 30/17		52	Nov 30/17
	7	Mar 25/21		53	Nov 30/17
	8	Dec 15/14		54	Nov 30/17
	9	Dec 15/14			
	10	Dec 15/14	91-34-40	1	Dec 15/14
	11	Dec 15/14		2	Dec 15/14
	12	Dec 15/14		3	Dec 15/14
	13	Dec 15/14		4	Nov 30/17
	14	Dec 15/14		5	Nov 30/17
	15	Dec 15/14		6	Nov 30/17
	16	Nov 30/17		7	Nov 30/17
	17	Nov 30/17		8	Nov 30/17
	18	Nov 30/17		9	Nov 30/17
	19	Nov 30/17		10	Sep 15/09
	20	Dec 15/14		11	Nov 30/17
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	24	Nov 30/17		15	Dec 15/14
	25	Nov 30/17		16	Nov 30/17
	26	Dec 15/14		17	Dec 15/14
	27	Nov 30/17		18	Dec 15/14
	28	Nov 30/17		19	Dec 15/14

CHAPTER 91

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
91-34-40 (cont.)	20	Sep 15/09	91-35-10	1	Nov 30/17
	21	Sep 15/09		2	Nov 30/17
	22	Nov 30/17		3	Nov 30/17
	23	Nov 30/17		4	Nov 30/17
	24	Nov 30/17		5	Nov 30/17
				6	Nov 30/17
91-34-50	1	Dec 15/14			
	2	Nov 30/17	91-35-20	1	Dec 15/14
	3	Nov 30/17		2	Dec 15/14
	4	Nov 30/17			
	5	Nov 30/17	91-37-20	1	Dec 15/14
	6	Nov 30/17		2	Nov 30/17
	7	Nov 30/17		3	Nov 30/17
	8	Sep 15/09		4	Nov 30/17
	9	Nov 30/17			
	10	Nov 30/17	91-52-70	1	Sep 15/09
	11	Sep 15/09		2	Sep 15/09
	12	Nov 30/17			
	13	Dec 15/14	91-73-30	1	Sep 15/09
	14	Nov 30/17		2	Dec 15/14
	15	Nov 30/17			
	16	Dec 15/14	91-74-20	1	Dec 15/14
	17	Nov 30/17		2	Dec 15/14
	18	Sep 15/09		3	Dec 15/14
	19	Nov 30/17		4	Nov 30/17
	20	Nov 30/17		5	Nov 30/17
	21	Dec 15/14		6	Dec 15/14
	22	Sep 15/09		7	Nov 30/17
	23	Sep 15/09		8	Dec 15/14
	24	Nov 30/17			
	25	Nov 30/17	91-77-10	1	Dec 15/14
	26	Dec 15/14		2	Dec 15/14
	27	Dec 15/14			
	28	Dec 15/14	91-77-20	1	Dec 15/14
	29	Nov 30/17		2	Dec 15/14
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	32	Nov 30/17		5	Dec 15/14
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	34	Nov 30/17		7	Nov 30/17
	35	Nov 30/17		8	Dec 15/14
	36	Nov 30/17			
	37	Nov 30/17			
	38	Nov 30/17			

CHAPTER 91

LIST OF EFFECTIVE PAGES (continued)

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
91-77-40	1	Dec 15/14			
	2	Dec 15/14			
	3	Nov 30/17			
	4	Nov 30/17			
	5	Nov 30/17			
	6	Nov 30/17			
	7	Dec 15/14			
	8	Nov 30/17			
	9	Nov 30/17			
	10	Nov 30/17			
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	21	Nov 30/17			
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	23	Nov 30/17			
	24	Nov 30/17			
	25	Nov 30/17			
	26	Nov 30/17			
	27	Nov 30/17			
	28	Nov 30/17			
	29	Nov 30/17			
	30	Nov 30/17			
91-79-30	1	Sep 15/09			
	2	Dec 15/14			
	3	Dec 15/14			
	4	Sep 15/09			

CHAPTER 91 - CHARTS AND WIRING DIAGRAMS

TABLE OF CONTENTS

S/N’s 4636776 and up, see also the PA-46-350P / PA-46R-350T Wiring Diagram Manual, P/N 767-116.

NOTE: Page number references used for the Wiring Diagrams (Schematics) above and on the following pages are coded.

The first six digits of the page number code are the Chapter, Section, and Sub-section and the remaining digits are the consecutive page number within that Sub-section - e.g., 91-28-10, page 23 would appear on these pages as “91281023” and 91-28-10, page 2 would appear as “9128102”.

CHAPTER 91 - CHARTS AND WIRING DIAGRAMS

TABLE OF CONTENTS (continued)

CHAPTER 91 - CHARTS AND WIRING DIAGRAMS

TABLE OF CONTENTS (continued)

CHAPTER 91 - CHARTS AND WIRING DIAGRAMS

TABLE OF CONTENTS (continued)

CHARTS

1. Torque Requirements

   Moved to 20-00-00.

2. Conversion Tables

   The following charts contain various conversion data that may be useful when figuring capacities, lengths, temperatures, and various weights and measures from the English system to the metric system or back again:

   Torque Conversion - “Chart 6” on page 91102

   Decimal Conversions - “Chart 7” on page 91103

   Temperature Conversion - “Chart 8” on page 91104

   Weights and Measures Conversion - “Chart 9” on page 91105

   Metric Conversion - “Chart 10” on page 91106

   Drill Sizes - “Chart 11” on page 91107

3. Hose Specifications

   See “Chart 12” on page 91108.

4. Consumable Materials

   A. General - See “Chart 13” on page 91109.

   B. Sealants - See “Chart 14” on page 911016

5. Vendor Contact Information

   See “Chart 15” on page 911019.

6. Electrical Wire Coding

   See “Chart 16” on page 911022.

7. Electrical Symbols

   See “Chart 17” on page 911025.

8. Electrical / Electronic Component Reference Designation Code

   See “Chart 18” on page 911027.

CHART 6
TORQUE CONVERSION

INCH POUNDS (IN.-LBS.) TO CENTIMETER KILOGRAMS (CMKG.)
CENTIMETER KILOGRAMS (CMKG.) TO INCH POUNDS (IN.-LBS.)
FOOT POUNDS (FT.-LBS.) TO METER KILOGRAMS (MKG.)
METER KILOGRAMS (MKG.) TO FOOT-POUNDS (FT.-LBS.)

IN.-LBS.	CMKG.
5	5.76
10	11.52
15	17.28
20	23.04
25	28.80
30	34.56
35	40.32
40	46.08
45	51.84
50	57.60
55	63.36
60	69.12
65	74.88
70	80.64
75	86.40
80	92.16
85	97.92
90	103.68
95	109.44
100	115.20
105	120.96
110	126.72
115	132.48
120	138.24

CMKG.	IN.-LBS.
50	43.4
100	86.8
150	130.2
200	173.6
250	217.0
300	260.4
350	303.8
400	347.2
450	390.6
500	434.0
550	477.4
600	520.8
650	564.2
700	607.6

FT.-LBS.	MKG.
2.5	.346
5	.691
7.5	1.037
10	1.383
12.5	1.728
15	2.074
17.5	2.419
20	2.765
22.5	3.111
25	3.456
27.5	3.802
30	4.148
32.5	4.493
35	4.839
37.5	5.185
40	5.530
42.5	5.876
45	6.222
47.5	6.567
50	6.913
52.5	7.258
55	7.604
57.5	7.950
60	8.295
62.5	8.641
65	8.987
67.5	9.332
70	9.678
72.5	10.024
75	10.369
77.5	10.715
80	11.060
82.5	11.406
85	11.752
87.5	12.097
90	12.443
92.5	12.789
95	13.134
97.5	13.480
100	13.826
105	14.517
110	15.208

FT.-LBS.	MKG.
115	15.900
120	16.591
125	17.282
130	17.974
135	18.665
140	19.356
145	10.047
150	20.739
155	21.430
160	22.121
165	22.813
170	23.504
175	24.195
180	24.887
185	25.578
190	26.269
195	26.960
200	27.652
105	28.343
210	29.034
215	29.726
220	30.417
225	31.108
230	31.800
235	32.491
240	33.182
245	33.873
250	34.565
255	35.256
260	35.947
265	36.639
270	37.330
275	38.021
280	38.713
285	39.404
290	40.095
295	40.786
300	41.478

MKG.	FT.-LBS.
1	7.23
2	14.46
3	21.69
4	28.98
5	36.16
6	43.39
7	50.63
8	57.86
9	65.09
10	72.32
11	79.56
12	86.79
13	94.02
14	101.26
15	108.49
16	115.72
17	122.95
18	130.19
19	137.42
20	144.65
21	151.89
22	159.12

CHART 7
DECIMAL CONVERSIONS

4ths	8ths	16ths	32nds	64ths	TO 3 PLACES	TO 2 PLACES	M.M. EQUIV
				1/64	.016	.02	.397
			1/32		.031	.03	.794
				3/64	.047	.05	1.191
		1/16			.062	.06	1.587
				5/64	.078	.08	1.984
			3/32		.094	.09	2.381
				7/64	.109	.11	2.778
	1/8				.125	.12	3.175
				9/64	.141	.14	3.572
			5/32		.156	.16	3.969
				11/64	.172	.17	4.366
		3/16			.188	.19	4.762
				13/64	.203	.20	5.159
			7/32		.219	.22	5.556
				15/64	.234	.23	5.593
1/4					.250	.25	6.350
				17/64	.266	.27	6.747
			9/32		.281	.28	7.144
				19/64	.297	.30	7.540
		5/16			.312	.31	7.937
				21/64	.328	.33	8.334
			11/32		.344	.34	8.731
				23/64	.359	.36	9.128
	3/8				.375	.38	9.525
				25/64	.391	.39	9.922
			13/32		.406	.41	10.319
				27/64	.422	.42	10.716
		7/16			.438	.44	11.112
				29/64	.453	.45	11.509
			15/32		.469	.47	11.906
				31/64	.484	.48	12.303
					.500	.50	12.700

4ths	8ths	16ths	32nds	64ths	TO 3 PLACES	TO 2 PLACES	M.M. EQUIV
				33/64	.516	.52	13.097
			17/32		.531	.53	13.494
				35/64	.547	.55	13.891
		9/16			.562	.56	14.288
				37/64	.578	.58	14.684
			19/32		.594	.59	15.081
				39/64	.609	.61	15.478
	5/8				.625	.62	15.875
				41/64	.641	.64	16.272
			21/32		.656	.66	16.669
				43/64	.672	.67	17.065
		11/16			.688	.69	17.462
				45/64	.703	.70	17.859
			23/32		.719	.72	18.256
				47/64	.734	.73	18.653
3/4					.750	.75	19.050
				49/64	.766	.77	19.447
			25/32		.781	.78	19.844
				51/64	.797	.80	20.241
		13/16			.812	.81	20.637
				53/64	.828	.83	21.034
			27/32		.844	.84	21.431
				55/64	.859	.86	21.828
	7/8				.875	.88	22.225
				57/64	.891	.89	22.622
			29/32		.906	.91	23.019
				59/64	.922	.92	23.416
		15/16			.938	.94	23.812
				61/64	.953	.95	24.209
			31/32		.969	.97	24.606
				63/64	.984	.98	25.003
					1.000	1.00	25.400

CHART 8
TEMPERATURE CONVERSION

CENTIGRADE - FAHRENHEIT

Example: To convert 20°C, to Fahrenheit, find 20 in the center column headed (°F - °C); then read 68.0°F, in the column (°F) to the right. To convert 20°F, to Centigrade; find 20 in the center column and read -6.67°C, in the (°C) column to the left.

°C	°F - °C	°F	°C	°F - °C	°F
-56.7	-70	-94.0	104.44	220	428.0
-51.1	-60	-76.0	110.00	230	446.0
- 45.6	-50	-58.0	115.56	240	464.0
- 40.0	-40	-40.0	121.11	250	482.0
- 34.0	-30	-22.0	126.67	260	500.0
- 28.9	-20	-4.0	132.22	270	518.0
- 23.3	-10	14.0	137.78	280	536.0
- 17.8	0	32.0	143.33	290	554.0
-12.22	10	50.0	148.89	300	572.0
-6.67	20	68.0	154.44	310	590.0
-1.11	30	86.0	160.00	320	608.0
4.44	40	104.0	165.56	330	626.0
10.00	50	122.0	171.11	340	644.0
15.56	60	140.0	176.67	350	662.0
21.11	70	158.0	182.22	360	680.0
26.67	80	176.0	187.78	370	698.0
32.22	90	194.0	193.33	380	716.0
37.78	100	212.0	198.89	390	734.0
43.33	110	230.0	204.44	400	752.0
48.89	120	248.0	210.00	410	770.0
54.44	130	266.0	215.56	420	788.0
60.00	140	284.0	221.11	430	806.0
65.56	150	302.0	226.67	440	824.0
71.00	160	320.0	232.22	450	842.0
76.67	170	338.0	237.78	460	860.0
82.22	180	356.0	243.33	470	878.0
87.78	190	374.0	248.89	480	896.0
93.33	200	392.0	254.44	490	914.0
98.89	210	410.0	260.00	500	932.0

CHART 9
WEIGHTS AND MEASURES CONVERSION

MULTIPLY	BY	TO OBTAIN
CENTIMETERS	0.3937	IN.
	0.03281	FT.
CU. CENTIMETERS	0.001	LITERS
	0.06102	CU. IN.
	0.0002642	U.S. GAL.
CU. FT.	28.320	CU. CM.
	1.728	CU. IN.
	7.481	U.S. GAL.
	28.32	LITERS
CU. IN.	16.39	CU. CM.
	0.01639	LITERS
	0.004329	U.S. GAL.
	0.01732	QUARTS
CU. METERS	1000000	CU. CM.
	35.314	CU. FT.
	61.023	CU. IN.
	264.17	GAL.
	999.97	LITERS
FEET	0.3048	METERS
	12.000	MILS
	304.8	MM.
	0.3333	YARDS
FT.-LB.	0.1383	M-KG
	0.001285	BTU
	0.000000376	KW-HR
FLUID OZ.	8	DRAM
	29.6	CU. CM.
GAL., IMPERIAL	277.4	CU. IN.
	1.201	U.S. GAL.
	4.546	LITERS
GAL., U.S. DRY	268.8	CU. IN.
	0.1556	CU. FT.
	1.164	U.S. GAL., LIQ.
	4.405	LITERS
GAL., U.S. LIQ.	231.0	CU. IN.
	0.1337	CU. FT.
	3.785	LITERS
	0.8327	IMPERIAL GAL.
	128	FLUID OZ.
IN.	2.540	CM.
	.08333	FT.
JOULES	0.000948	BTU
	0.7376	FT.-LB.

MULTIPLY	BY	TO OBTAIN
KILOGRAMS	2.205	LB.
	35.27	OZ.
	1000	GRAMS
LITERS	1000	CU. CM.
	61.03	CU. IN.
	0.03532	CU. FT.
	0.2642	U.S. GAL.
	0.22	IMPERIAL GAL.
	1.057	QUARTS
METERS	39.37	IN.
	3.281	FT.
	1000	MM.
METER-KILOGRAM	7.233	FT.-LB.
	9.807	JOULES
OUNCES, AVDP	0.0625	LB., AVDP
	28.35	GRAMS
	437.5	GRAINS
OUNCES, FLUID	29.57	CU. CM.
	1.805	CU. IN.
LB., AVDP	453.6	GRAMS
	7000	GRAINS
	16.0	OUNCES
SQUARE INCH	6.4516	SQ. CM.
POUND PER SQUARE INCH (PSI)	0.0703	KG.-CM SQUARED
STATUTE MILE	1.609	KILOMETER
	0.8684	NAUTICAL MILE
NAUTICAL MILE	1.151	STATUTE MILE
QUART	.9463	LITER
MILLIMETER	1000	MICRON
MICRON	0.001	MILLIMETER
	0.000039	INCH
INCH POUNDS	11.521	METER GRAMS
INCH OUNCES	0.72	METER GRAMS
POUNDS	0.453	KILOGRAMS

CHART 10
METRIC CONVERSION

Example: Convert 1.5 inches to millimeters.

(1) Read down inches column to 1. inches.
(2) Read across top inch column to 0.5.
(3) Read down and across to find millimeters (1.5 inches is 38.10 millimeters).

INCHES TO MILLIMETER

INCHES	0.0000	0.0001	0.0002	0.0003	0.0004	0.0005	0.0006	0.0007	0.0008	0.0009
					MILLIMETER					
0.000		0.0025	0.0050	0.0076	0.0101	0.0127	0.0152	0.0177	0.0203	0.0228
0.001	0.0254	0.0279	0.0304	0.0330	0.0355	0.0381	0.0406	0.0431	0.0457	0.0482
0.002	0.0508	0.0533	0.0558	0.0584	0.0609	0.0635	0.0660	0.0685	0.0711	0.0736
0.003	0.0762	0.0812	0.0838	0.0863	0.0889	0.0914	0.0939	0.0965	0.0965	0.0990
0.004	0.1016	0.1041	0.1066	0.1092	0.1117	0.1143	0.1168	0.1193	0.1219	0.1244
0.005	0.1270	0.1295	0.1320	0.1346	0.1371	0.1397	0.1422	0.1447	0.1447	0.1498
0.006	0.1524	0.1549	0.1574	0.1600	0.1625	0.1651	0.1676	0.1701	0.1727	0.1752
0.007	0.1778	0.1803	0.1828	0.1854	0.1879	0.1905	0.1930	0.1955	0.1981	0.2006
0.008	0.2032	0.2057	0.2082	0.2108	0.2133	0.2159	0.2184	0.2209	0.2235	0.2260
0.009	0.2286	0.2311	0.2336	0.2362	0.2387	0.2413	0.2438	0.2463	0.2489	0.2514
INCHES	0.000	0.001	0.002	0.003	0.004	0.005	0.006	0.007	0.008	0.009
					MILLIMETER					
0.00		0.025	0.050	0.076	0.101	0.127	0.152	0.177	0.203	0.228
0.01	0.254	0.279	0.304	0.330	0.355	0.381	0.406	0.431	0.457	0.482
0.02	0.508	0.533	0.558	0.584	0.609	0.635	0.660	0.685	0.711	0.736
0.03	0.762	0.787	0.812	0.838	0.863	0.889	0.914	0.939	0.965	0.990
0.04	1.016	1.041	1.066	1.092	1.117	1.143	1.168	1.193	1.219	1.244
0.05	1.270	1.295	1.320	1.346	1.371	1.397	1.422	1.447	1.473	1.498
0.06	1.524	1.549	1.574	1.600	1.625	1.651	1.676	1.701	1.727	1.752
0.07	1.778	1.803	1.828	1.854	1.879	1.905	1.930	1.955	1.981	2.006
0.08	2.032	2.057	2.082	2.108	2.133	2.159	2.184	2.209	2.235	2.260
0.09	2.286	2.311	2.336	2.362	2.387	2.413	2.438	2.463	2.489	2.514
INCHES	0.00	0.01	0.02	0.03	0.04	0.05	0.06	0.07	0.08	0.09
					MILLIMETER					
0.0		0.254	0.508	0.762	0.016	1.270	1.524	1.778	2.032	2.286
0.1	2.540	2.794	3.048	3.302	3.556	3.810	4.064	4.318	4.572	4.826
0.2	5.080	5.334	5.558	5.842	6.096	6.350	6.604	6.858	7.112	7.366
0.3	7.620	7.874	8.128	8.382	8.636	8.890	9.144	9.398	9.652	9.906
0.4	10.160	10.414	10.668	10.922	11.176	11.430	11.684	11.938	12.192	12.446
0.5	12.700	12.954	13.208	13.462	13.716	13.970	14.224	14.478	14.732	14.986
0.6	15.240	15.494	15.748	16.002	16.256	16.510	16.764	17.018	17.272	17.526
0.7	17.780	18.034	18.288	18.542	18.796	19.050	19.304	19.558	19.812	20.066
0.8	20.320	20.574	20.828	21.082	21.336	21.590	21.844	22.098	22.352	22.606
0.9	22.860	23.114	23.368	23.622	23.876	24.130	24.384	24.638	24.892	25.146
INCHES	0.00	0.1	0.2	0.3	0.4	0.5	0.6	0.7	0.8	0.9
					MILLIMETER					
0.0		2.54	5.08	7.62	10.16	12.70	15.24	17.78	20.32	22.86
1.0	25.40	27.94	30.48	33.02	35.56	38.10	40.64	43.18	45.72	48.26
2.0	50.80	53.34	55.88	58.42	60.96	63.50	66.04	68.58	71.12	73.66
3.0	76.20	78.74	81.28	83.82	86.36	88.90	91.44	93.98	96.52	99.06
4.0	101.60	104.14	106.68	109.22	111.76	114.30	116.84	119.38	121.92	124.46
5.0	127.00	129.54	132.08	134.62	137.16	139.70	142.24	144.78	147.32	149.86
6.0	152.40	154.94	157.48	160.02	162.56	165.10	167.64	170.18	172.72	175.26
7.0	177.80	180.34	182.88	185.42	187.96	190.50	193.04	195.58	198.12	200.66
8.0	203.20	205.74	208.28	210.82	213.36	215.90	218.44	220.98	223.52	226.06
9.0	228.60	231.14	233.68	236.22	238.76	241.30	243.84	246.38	248.92	251.46

CHART 11
DRILL SIZES

Decimal/Millimeter Equivalents of Drill Sizes From 1/2" to No. 80

Size	Decimal Equiv.	Millimeter Equiv.	Size	Decimal Equiv.	Millimeter Equiv.	Size	Decimal Equiv.	Millimeter Equiv.	Size	Decimal Equiv.	Millimeter Equiv.
1/2	0.500	12.7000	G	0.261	6.6294	5/32	0.1562	3.9687	51	0.067	1.7018
31/64	0.4843	12.3031	F	0.257	6.5278	23	0.154	3.9116	52	0.0635	1.6129
15/32	0.4687	11.9062	E-1/4	0.250	6.3500	24	0.152	3.8608	1/16	0.0625	1.5875
29/64	0.4531	11.5094	D	0.246	6.2484	25	0.1495	3.7973	53	0.0595	1.5113
7/16	0.4375	11.1125	C	0.242	6.1468	26	0.147	3.7338	54	0.055	1.397
27/64	0.4218	10.7156	B	0.238	6.0452	27	0.144	3.6576	55	0.052	1.3208
Z	0.413	10.4902	15/64	0.2343	5.9531	9/64	0.1406	3.5719	3/64	0.0468	1.1906
13/32	0.4062	10.3187	A	0.234	5.9436	28	0.1405	3.5687	56	0.0465	1.1811
Y	0.404	10.2616	1	0.228	5.7912	29	0.136	3.4544	57	0.043	1.0922
X	0.397	10.0838	2	0.221	5.6134	30	0.1285	3.2639	58	0.042	1.0668
25/64	0.3906	9.9212	7/32	0.2187	5.5562	1/8	0.125	3.1750	59	0.041	1.0414
W	0.386	9.8044	3	0.213	5.4102	31	0.120	3.048	60	0.040	1.016
V	0.377	9.5758	4	0.209	5.3086	32	0.116	2.9464	61	0.039	0.9906
3/8	0.375	9.5250	5	0.2055	5.2197	33	0.113	2.8702	62	0.038	0.9652
U	0.368	9.3472	6	0.204	5.1816	34	0.111	2.8194	63	0.037	0.9398
23/64	0.3593	9.1262	13/64	0.2031	5.1594	35	0.110	2.794	64	0.036	0.9144
T	0.358	9.1281	7	0.201	5.1054	7/64	0.1093	2.7781	65	0.035	0.899
S	0.346	8.7884	8	0.199	5.0546	36	0.1065	2.7051	66	0.033	0.8382
11/32	0.3437	8.7300	9	0.196	4.9784	37	0.104	2.6416	1/32	0.0312	0.7937
R	0.339	8.6106	10	0.1935	4.9149	38	0.1015	2.5781	67	0.032	0.8128
Q	0.332	8.4328	11	0.191	4.8514	39	0.0995	2.5273	68	0.031	0.7874
21/64	0.3281	8.3337	12	0.189	4.8006	40	0.098	2.4892	69	0.029	0.7366
P	0.323	8.2042	3/16	0.1875	4.7625	41	0.096	2.4384	70	0.028	0.7112
O	0.316	8.0264	13	0.185	4.699	3/32	0.0937	2.3812	71	0.026	0.6604
5/16	0.3125	7.9375	14	0.182	4.6228	42	0.0935	2.3749	72	0.025	0.635
N	0.302	7.6708	15	0.180	4.572	43	0.089	2.2606	73	0.024	0.0696
19/64	0.2968	7.5387	16	0.177	4.4958	44	0.086	2.1844	74	0.0229	0.58166
M	0.295	7.4930	17	0.173	4.3942	45	0.082	2.0828	75	0.021	0.5334
L	0.290	7.3660	11/64	0.1718	4.3656	46	0.081	2.0574	76	0.020	0.508
9/32	0.2812	7.1425	18	0.1695	4.3053	47	0.0785	1.9939	77	0.018	0.4572
K	0.281	7.1374	19	0.166	4.2164	5/64	0.0781	1.9844	1/64	0.0156	0.3969
J	0.277	7.0358	20	0.161	4.0894	48	0.076	1.9304	78	0.016	0.4064
I	0.272	6.9088	21	0.159	4.0386	49	0.073	1.8542	79	0.0145	0.3683
H	0.266	6.7564	22	0.157	3.9878	50	0.070	1.778	80	0.0135	0.3429
17/64	0.2656	6.7462									

DRILL SIZES AVAILABLE

Drill may be obtained in regular sizes to a 4 inch diameter, and increase in 64ths of an inch. The regular metric drills vary from 2 to 76mm and increase in 0.5mm variations.

CHART 12
HOSE SPECIFICATIONS

SINGLE WIRE BRAID FABRIC COVERED

MIL PART NO.	TUBE SIZE O.D.	HOSE SIZE I.D.	HOSE SIZE O.D.	RECOMMENDED OPER. PRESS	MIN BURST PRESS	MAX PROOF PRESS	MIN BEND RADIUS
MIL-H-8794- 3-L	3/16	1/8	.45	3,000	12,000	6,000	3.00
MIL-H-8794- 4-L	1/16	3/16	.52	3,000	12,000	6,000	3.00
MIL-H-8794- 5-L	5/16	1/4	.58	3,000	10,000	5,000	3.38
MIL-H-8794- 6-L	3/8	5/16	.67	2,000	9,000	4,500	4.00
MIL-H-8794- 8-L	1/2	13/32	.77	2,000	8,000	4,000	4.63
MIL-H-8794-10-L	5/8	1/2	.92	1,750	7,000	3,500	5.50
MIL-H-8794-12-L	3/4	5/8	1.08	1,500	6,000	3,000	6.50
MIL-H-8794-16-L	1	7/8	1.23	800	3,200	1,600	7.38
MIL-H-8794-20-L	1 1/4	1 1/8	1.50	600	2,500	1,250	9.00
MIL-H-8794-24-L	1 1/2	1 3/8	1.75	500	2,000	1,000	11.00
MIL-H-8794-32-L	2	1 13/16	2.22	300	1,400	700	13.25
MIL-H-8794-40-L	2 1/2	2 3/8	2.88	200	1,000	300	24.00
MIL-H-8794-48-L	3	3	3.56	200	800	300	33.00

Construction: Seamless synthetic rubber inner tube reinforced with one fiber braid, one braid of high tensile steel wire and covered with an oil resistant rubber impregnated fiber braid.

Identification: Hose is identified by specification number, size number, quarter year and year, hose manufacturer's identification.

Uses: Hose is approved for use in aircraft hydraulic, pneumatic, coolant, fuel and oil systems.

Operating Temperatures:

Sizes -3 thru -12: Minus 65°F. to plus 250°F.

Sizes -16 thru -48: Minus 40°F to plus 275°F.

NOTE: Maximum temperatures and pressures should not be used simultaneously.

MULTIPLE WIRE BRAID RUBBER COVERED

MIL PART NO.	TUBE SIZE O.D.	HOSE SIZE I.D.	HOSE SIZE O.D.	RECOMMENDED OPER. PRESS	MIN BURST PRESS	MAX PROOF PRESS	MIN BEND RADIUS
MIL-H-8788- 4-L	1/4	7/32	.63	3,000	16,000	8,000	3.00
MIL-H-8788- 5-L	5/16	9/32	.70	3,000	14,000	7,000	3.38
MIL-H-8788- 6-L	3/8	11/32	.77	3,000	14,000	7,000	5.00
MIL-H-8788- 8-L	1/2	7/16	.86	3,000	14,000	7,000	5.75
MIL-H-8788-10-L	5/8	9/16	1.03	3,000	12,000	6,000	6.50
MIL-H-8788-12-L	3/4	11/16	1.22	3,000	12,000	6,000	7.75
MIL-H-8788-16-L	1.00	7/8	1.50	3,000	10,000	5,000	9.63

Hose Construction: Seamless synthetic rubber inner tube reinforced with one fabric braid, two or more steel wire braids, and covered with a synthetic rubber cover (for gas applications, request perforated cover).

Identification: Hose is identified by specification number, size number, quarter year and year, hose manufacturer's identification.

Uses: High pressure hydraulic, pneumatic, coolant, fuel and oil.

Operating Temperature:

Minus 65°F to plus 200°F.

CHART 13 (Sheet 1 of 7)
CONSUMABLE MATERIALS - GENERAL

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
ABS-Solvent/ Cements		Solarite, #11 Series	Solar Compounds Corp.
Adhesive		EC 801 EC 807 EC 1357 Scotch Grip 210 (Rubber Adhesive)	Minnesota Mining and Manufacturing (3M) Adhesive Coating and Sealers Division
		Hysol EA 9446 (Piper P/N 279-129)	Hensel Corp.
Adhesive, Deice Boot		EC-1300L (Piper P/N 179-929)	3M
Adhesive, Fuel Barrier		A-851-B (Piper P/N 279-137)	Goodrich
Anti-Galling Solution	MIL-PRF-907	Ease-Off	Taxacone Company
Anti-Seize Compound (Graphite Petrolatum)	SAE-AMS-2518	Armite Product	Armite Laboratories
		Anti-Seize Compound	Exxon Oil Company
		Royco 44	Royal Lubricants Co.
Anti-Seize Compound	TT-A-580 (TT-S-1732)	Armite Product	Armite Laboratories
Anti-Seize Thread Compound “HIGH TEMPERATURE”		Fel-Pro C5-A	Fel-Pro Incorporated
Cleaner and Polish, Plexiglas	P-P-560	Part Number 403D	Permatex Co., Inc. Kansas City, Kansas 66115
Cleaners, General		Fantastic Spray Perchlorethylene VM&P Naphtha (Lighter Fluid)	Local Supplier
Cloth, Clean	SAE ASM 3819	Cloth, Class 1 or 2 Grade A (Piper P/N 280-686)	Local Supplier
Compound, Buffing and Rubbing		Automotive Type Axalta (formerly DuPont) #7	Axalta Coating Systems
		Ram Chemical #69	Ram Chemicals
Compound, Polishing		Mirror Glaze	Mirror Bright Polish

CHART 10 (Sheet 2 of 7)
CONSUMABLE MATERIALS - GENERAL

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
			Co., Incorporated
Corrosion Retardant Compounds	MIL-PRF-16173E	LPS-3 Heavy Duty Rust Inhibitor (Piper P/N 197-508)	LPS Laboratories
		Metal Parts Protector Protector Flex (Piper P/N 197-509)	Chemi-Cap. Chemical Packaging Corp.
Deicer Boot Surface Coatings		Agemaster	B.F. Goodrich
Dry Lubricant		MS-122AD	Miller-Stephenson
Gasket Cement		Permatex No. 2	Permatex Company, Inc.
Grease, Actuator		2196-74-1	Dukes Astronautics Co.
Grease, Aircraft Instrumentation, Gear and Actuator Screw (-73 to +121°C) (-94 to +250°F)	MIL-PRF-23827C (See Note at end.)	Supermil Grease No. A72832	Amoco
		Royco 27A	Royal Lubricants Co.
		Shell 6249 Grease	Shell Oil Company
		RR-28	Socony Mobil Oil Co.
		Castrolease A1	Burmah-Castrol LTD.
		Low-Temp. Grease E.P.	Texaco Incorp.
		5114 E.P. Grease AV55	Standard Oil of Calif.
		Aeroshell Grease 7 Braycote 627S	Shell Oil Company
		Mobil Grease 33 (755-162 (12.5 oz. Tube))	Mobil Oil Corporation
		B.P. Aero Grease 31B	B.P. Trading Limited
Grease, Ball and Roller Bearing	DOD-G-24508	Regal ASB-2 Formula TG-10293	Texaco Incorporated
		Andok B Code 1-20481, Darina Grease 1 XSG-6213 Code 71-501, Darina Grease 2 XSG-6152 Code 71-502, Alvania Grease 2 XSG-6151 Code 71-012, Cyprina Grease 3 XSG-6280	Exxon Company, U.S.A. Shell Oil Company

CHART 10 (Sheet 3 of 7)
CONSUMABLE MATERIALS - GENERAL

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
		Code 71-003	
Grease, General Purpose Wide Temperature	MIL-PRF-81322G	Marfax All Purpose	Texaco Incorporated
		Aeroshell No. 6	Shell Oil Company
(-54 to +177 °C) (-65 to +350 °F)		Mobil Grease 77 or Mobilux EP2	Mobil Oil Corporation
		Shell Alvania EP2	Shell Oil Company
		Royco 22	Royal Lubricants Company
		Mobil Grease 28	Mobil Oil Corporation
		Aeroshell No. 22	Shell Oil Company
Grease, Lubricating, Molybdenum Disulfide, Low and High Temperature	MIL-G-21164	Aeroshell Grease No. 17	Shell Oil Company
		Royco 64C	Royal Lubricants Co.
		Castrolease MSA (c)	Burmah Castrol LTD.
Grease, Lubricating, Plug Valve, Gasoline and Oil Resistant	SAE-AMS-G-6032	Royco 32	Royal Lubricant Co.
		Castrolease PV	Burmah Castrol LTD.
		Parker Fuel Lube 44	Parker Seal Company
		B.P. Aero Grease 32	B.P. Trading Limited
		L-237	Lehigh Tenneco Chemicals Co., Inc.
		Rockwell 950	Rockwell International
Grease, Waterproof, High and Low Temperature		Lubriplate Aero (Piper P/N 197-515. 5 Gal.)	LUBRIPLATE Lubricants Company
“Hot Melt” Adhesive Polyamids and “Hot Melt” Gun.	Stick Form 1/2 in. diameter, 3 in. long		Sears, Roebuck and Company or most hardware stores.

CHART 10 (Sheet 4 of 7)
CONSUMABLE MATERIALS - GENERAL

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
Hydraulic Fluid	MIL-PRF-5606	Brayco 756D	Bray Oil Company
		TL-5874	Texaco Incorporated
		PED 3565	Standard Oil Company of California
		Aircraft Hydraulic Oil AA	Texaco Incorporated
		RPM Aviation Oil No. 2 Code PED 2585 PED 3337	Standard Oil Company of California
		Aeroshell Fluid 4, SL-7694	Shell Oil Company
		Aero HF	Mobil Oil Corporation
		Royco 756, 756A and 756B	Royal Lubricants Co.
Isopropyl Alcohol	Fed. Spec. TT-I-735		Local Supplier
Isocryl Tape		(PMS-C1012-2)	Schnee Moorehead Chemicals, Incorporated
Kevlar		Kevlar	Kevlar Special Products
Leak Detector Solution for Oxygen Systems	MIL-PRF-25567	ALPHA 73 Oxygen Leak Detector Type 1	U.S. Gulf Corporation
		Leak Tec #16-OX	American Gas and Chemical Co. LTD.
Loctite	ASTM-D-5363	Loctite 290 (Red)	Loctite Corporation
		Loctite 222 (Brown)	
		Loctite 27121 (10 ml) (Piper P/N 279-128)	
Masking Tape 1”		Scotch No. 2308 or equivalent	3M
Methylethylketone	Fed. Spec. TT-M-261		Local Supplier
Molybdenum Disulfide	SAE-AMS-M-7866	Molykote-Type G (Paste)	Dow Corning Corp.
		Molykote - Type 2 (Powder)	

CHART 10 (Sheet 5 of 7)
CONSUMABLE MATERIALS - GENERAL

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
Oil, Air Conditioner, R12		Frigidaire #525	Virginia Chemical
		Suniso #5	Sun Oil Company of Pennsylvania
		Texaco Capilla “E”	Texaco Incorporated
Oil, Air Conditioner, HFC-134a, POE		Ester-25065 (Piper P/N 197-511)	
Oil Lubricating, General Purpose, Low Temperature	MIL-PRF-7870C	Caltex Low Temp. Oil	Caltex Oil Products Company
		Sinclair Aircraft Orbit Lube	Sinclair Refining Company
		1692 Low Temp Oil	Texaco Incorporated
		Aviation Instrument Oil	Standard Oil Company of California
		Royco 363	Royal Lubricants Co.
Pad, Abrasive, Type A		3M Scotch-Brite 7447 (Piper P/N 279-235)	3M
Patching Compound Epoxy		Solarite #400	Solar Compounds Corp.
Primer, Fluid Resistant Epoxy		EWDE072A/B (Piper P/N 279-179)	PPG Aerospace PRC-DeSoto
		10P8-10NF / EC-283 (Piper P/N 279-506)	Akzo Nobel Aerospace Coatings
		10P30-5 / EC-275 (Piper P/N 279-108)	
		44GN036	Deft, Inc.
		Axalta (formerly Dupont) Expoxy 13550S (Piper P/N 279-240)	Axalta Coating Systems
Rain Repellent	FSCM 50150	Repcon	Unelco Corporation
Refrigerant, A/C HFC-134a	1,1,1,2-Tetrafluoroethane and/or CAS# 811-97-2		Procure locally.
Safety Walk Pressure Sensitive		Flextred 300	Wooster Products, Incorporated

CHART 10 (Sheet 6 of 7)
CONSUMABLE MATERIALS - GENERAL

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
Sealant	See "Chart 14" on page 911016		
Sealer		PR 1321 B-1/2	Products Research Company
Sealer, Deice Boot		A-56-B-R1 (Piper P/N 279-142)	Goodrich
Sealing Compound, Gasket and Joint		Tite-Seal	Radiator Specialty Co.
Silicone Compound -54 to +204 °C -65 to +400 °F	SAE-AS-8660	DC-4, DC-6 Compound (Piper P/N 279-149)	Dow Corning
		G-624	General Electric Co. Silicone Products Department
Solvents		Methylethyl Ketone	Local Suppliers
		Methylene Chloride	
	ASTM D329	Acetone (Piper P/N 193-035)	
		Y2900	Union Carbide; Plastic Division
	Fed. Spec. PD 680 Type I - Stoddard Solvent		Local Supplier
	Type II - High Temperature		Local Supplier
Solvent, Cleaning, Propeller Slip Ring		CRC-2-26	Corrosion Reaction Consultants, Inc.
Thinner (Toulene)		ADC 108-88-3 (Piper P/N 179-451)	Ashland
Teflon Tape	.003" x .5" wide/-1		Minnesota Mining and Manufacturing Company
			Shamban W.S. and Co.
	.003" x .25" wide/-2		Johnson & Johnson, Inc. Permacel Division

CHART 10 (Sheet 7 of 7)
CONSUMABLE MATERIALS - GENERAL

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
Thread Lubricant, Oleo Strut, Air Valve	MIL-PRF-907E	Kopr-Kote	Jet Lube, Inc.
Thread Sealant for High Pressure Oxygen System	A-A-58092	Permacel 412	Johnson & Johnson, Inc. Permacel Division
Toluol	TT-M-261		Local Supplier
Trichlorethylene	MIL-T-7003	Perm-A-Clor	Dextrex Chemical Industries, Inc.
		Turco 4217	Turco Products, Inc.
Vinyl Foam	1 in. x 1/8 in.	530 Series, Type I	Norton Tape Division
Vinyl, Foam Tape	1/8 in. x 1 in.	501 Series, Type II	Norton Tape Division
Vinyl, Black Plastic	2 in. x 9 mil. and/or 1 1/2 in. x 9 mil.		

NOTE: Take precautions when using MIL-PRF-23827 and engine oil. These lubricants contain chemicals harmful to painted surfaces.

CHART 14 (Sheet 1 of 3)
CONSUMABLE MATERIALS - SEALANTS

Sealant Type	Manufacturer Product (Specific Gravity & Cured Hardness)	Class/ Worklife (hrs)	Typical Time at 77°F/50%RH		Piper Code			Shelf Life at 80°F (months)
			Tack Free	Cure to 30A	2.5-oz Semco	6-oz Semco	Gallon Can Kit	
Type 1 **Fuel Tank and Fuselage Pressurized Structure** See Note 1 and see also Type 6.	3M AC -236 (SG=1.64, ≥60A)	A-2	24	48	279-554	279-060	279-514	9
		B-½	8	24	279-483	279-516	---	9
		B-2	24	48	279-555	279-517	---	9
		B-4	48	90	279-556	279-066	279-515	9
	3M AC -350 (SG=1.40, ≥48A)	A-2	8	12	279-518	279-185	279-524	9
		B-½	2-3	2-3	279-519	279-186	---	9
		B-2	7-8	7-8	279-520	279-187	---	9
		B-4	32-36	32-36	279-521	279-183	279-525	9
	Flamemaster CS3204 (SG=1.66, ≥55A)	A-2	24	72	279-526	279-213	279-532	9
		B-½	8	30	279-527	279-217	---	9
		B-2	24	72	279-528	279-212	---	9
		B-4	36	90	279-529	279-220	279-533	9
	Royal WS -8020 (SG=1.52, ≥50A)	A-2	<24	<48	279-822	279-821	279-810	9
		B-½	<10	<24	279-534	279-557	---	9
		B-2	<24	<48	279-535	279-548	---	9
		B-4	<36	<90	279-823	279-824	279-807	9
	Royal WS-8020RC (SG=1.45, ≥50A)	A-2	<10	<30	279-536	179-202	179-209	9
		B-½	<5	<5	279-537	179-203	---	9
		B-2	<9	<14	279-538	179-204	---	9
		B-4	<24	<26	279-539	179-205	179-210	9
	PPG PR -1440 (SG=1.57, ≥46A)	A-2	<36	72	279-540	279-545	279-544	9
		B-½	<4	8	279-541	279-546	---	9
		B-2	<8	11	279-542	279-122	---	9
Type 1LD **Fuel Tank and Fuselage Pressurized Structure** See Note 2	Royal WS -8032 (SG=1.10, ≥48A)	B-½	<4	<6	279-428	279-432	---	9
		B-2	<6	<9	279-429	279-433	---	9
	PPG PR -2007 (SG=1.10, ≥45A)	B-½	4	6	279-426	279-430	---	9
		B-2	<6	8	279-427	279-431	---	9

(PIR-PPS45012-1 Rev. AU.)

(1) Use Accelerator below if desired.
(2) Low Density. Use Accelerator below if desired.

CHART 11 (Sheet 2 of 3)
CONSUMABLE MATERIALS - SEALANTS

Sealant Type	Manufacturer Product & Class (Specific Gravity & Cured Hardness)	Worklife/ Assembly Time (hrs)	Typical Time at 77°F/50%RH		Piper Code	Shelf Life at 80°F (months)
			Tack Free	Cure to 30A		
Type 2 Access Doors and Panels in Pressurized Structure See Note 3	3M AC-215 B-2 (SG=1.52, 45A)	2	24	48	279-184	12
	Flamemaster CS3330ci B-2 (SG=1.50, 35A)	2	18	30	279-820	9
	PPG PR-1428 B-2 (SG=1.50, 50A)	2	<8	24	279-036	12
	Royal WS8010 B-1/2 (SG=1.50, 53A)	30 min	<3	<4	279-497	9
	Royal WS8010 B-2 (SG=1.50, 53A)	2	<6	<12	279-496	9
Type 3 Fill and Drain Coating in Small Tanks	3M EC-776SR (SG=0.86)	-	20 min	24	279-229	6
	Flamemaster CS3600 (SG=0.88)	-	20 min	24-48	279-547	12
	PPG PR-1005-L (SG=0.85)	-	20 min	48	179-750	12
Type 4, Group I General Purpose Silicone See Note 4	Momentive RTV102 -White (SG=1.05, 30A)	-	20 min	24	179-758	12
	Momentive RTV103 -Black (SG=1.05, 30A)	-	20 min	24	179-759	12
	Momentive RTV108 -Clear (SG=1.05, 30A)	-	20 min	24	179-760	12
Type 4, Group III Engine Compartment Silicone See Note 5	Momentive RTV106 -Red (SG=1.07, 30A)	-	20 min	24	179-763	12
Type 5 **Wet Fastener Installation in Pressurized Structure.** See Note 6	3M AC-236 C-20 (SG=1.64, 60A)	8(20)	60	<336	279-196	9
	3M AC-350 C-8(24) (SG=1.40, 50A)	8(24)	96	96	279-523	9
	Royal WS -8032 C-8 (SG=1.17, 50A)	8(24)	<168	<168	279-550	9
	PPG P/S 890 C-8(20) (SG=1.54, 45A)	8(20)	<168	<168	279-531	9
	PPG PR-1440 C-20 (SG=1.57, 46A)	8(20)	<168	<168	279-192	9
	PPG PR-1776 C-8 (SG=1.29, 48A)	8(24)	<120	120	279-551	9
Type 6 Fuel Tank and Fuselage Pressurized Structure See Note 7	3M AC-250 A-1/6 (SG=1.61, 55A)	10 min	70 min	2	279-244	6
	3M AC-250 B-1/6 (SG=1.62, 55A)	10 min	1	2	279-245	6
	PPG PS 860 B-1/6 (SG=1.65, 45A)	20 min	<3	8	279-191	6
Type 7 **Firewall** See Note 8	Cytec DAPCO 2100 (SG=1.37, 50A)	10 min	15 min	24	279-522	6
	Cytec DAPCO 2200 (SG=1.37, 50A)	30 min	1	4	279-530	6
	Esterline Fastblock 100 (SG=1.25, 63A)	4	6	<48	279-543	6
	Flamemaster CS1900 (SG=1.33, 80A)	2	<4	<60	279-226	12
	Momentive RTV133 (SG=1.23, 45A)	1	3-5	48	279-029	12
	PPG PR-812 (SG=1.33, 75A)	2	<24	48	279-549	12

(3) Low Adhesion.
(4) Minimum exposure to fuel, not paintable.
(5) Minimal exposure to fuel, intermittent temperature up to to 600°F not paintable.
(6) Use Accelerator below if desired. AMS specifications for class C sealants contain no standard cure requirements.
(7) Short Work Life Repairs.
(8) 15 minutes at 2,000 °F.

CHART 11 (Sheet 3 of 3)
CONSUMABLE MATERIALS - SEALANTS

Sealant Type	Manufacturer Product & Class (Specific Gravity & Cured Hardness)	Worklife/ Assembly Time (hrs)	Typical Time at 77°F/50%RH		Piper Code	Shelf Life at 80°F (months)
			Tack Free	Cure to 30A		
Type 8 Pressurized Windows or Deice Boot Edge Sealer	Flamemaster CS3247 B-4 (SG=1.63, 45A)	4	<36	<90	279-227	9
	PPG PR-1425CF B-2 (SG=1.44, 47A)	2	<8	12	279-019	9
	Flamemaster CS3247 B-2 (SG=1.63, 45A)	2	<24	48	279-560	9
Type 9 Accelerator See Note 9	PPG PR-716 Turquoise Liquid	-	<30 min	-	279-816	9
	PPG PR-717 Green Liquid	-	<30 min	-	279-552	9
	PPG P/S 815 Clear Aerosol	-	<2	-	279-553	6

(9) See above for applicability.

CHART 15 (Sheet 1 of 3)
VENDOR CONTACT INFORMATION

A

American Gas and
Chemical Co. LTD
220 Pegasus Avenue
Northvale, NJ 07647
201-767-7300

Amoco Oil Co.
200 E. Randolph Drive
Chicago, IL 60601
312-856-5111

Armite Laboratories
1845-49 Randolph Street
Los Angeles, CA 90001
213-587-7744

Akzo Nobel Aerospace Coatings
East Water Street
Waukegan, IL 60085
847-625-3340
http://www.anac.com/

Axalta Coating Systems
Suite 3600
2001 Market Street
Philadelphia, PA 19103 USA
855-547-1461
http://www.axaltacs.com

B

Bostik, Inc.
211 Boston Street
Middleton, MA 01949
888-603-8558

BP Trading Limited
Moore Lane
Brittanic House
London E.C. 2
England

Bray Oil Company
1925 N. Marianna Avenue
Los Angeles, CA 98103
213-268-6171

Burmah - Castrol Inc.
30 Executive Avenue
Edison, NJ 08817
201-287-3140

C

California Texas Oil Corp.,
380 Madison Avenue
New York, NY 10017

Caltex Oil Products Co.
New York, NY 10020

CEE BEE Chemical Co.
9520 E. CEE BEE Drive
Box 400
Downey, CA 92041

Chemi-cap
Chemical Packaging Corp.
1100 N.W. 70th Street
Ft. Lauderdale, FL 33309
305-665-9059

Corrosion Reaction
Consultants, Inc.
Limekin Pike
Dresher, PA 19025

D

Deft, Inc.
17451 Von Karman Ave.
Irvine, CA 92614
800-544-3338
http://www.deftfinishes.com/

Dextrex Chemical
P. O. Box 501
Detroit, MI 48232

Dow Corning Corporation
Alpha Molykote Plant
64 Harvard Avenue
Stanford, CT 06902

Dukes Astronautics Co.
7866 Deering Avenue
Canoga Park, CA 91304

DuPont Company
Finishes Div.
DuPont Building
Wilmington, DE 19898
302-774-1000

E

Exxon Oil Company
1251 Avenue of the Americas
New York, NY 10020
212-398-3093

F

Fel-Pro Incorporated
7450 N. McCormick Blvd.
Box C1103
Skokie, IL 60076
312-761-4500

G

General Electric Co.
Silicone Products Dept.
Waterford, NY 12188
518-237-3330

Guardian Avionics
1840 E Valencia Rd Suite 216
Tucson, AZ. 85706
(520-889-1177
https://www.guardianavionics.com/recalibration

H

Hensel Corp., Aerospace Group
2850 Willow Pass Road
P.O. Box 312
Bay Point, CA 94565
925-458-8030

J

Jet Lube, Inc.
P.O. Box 21258
Houston, TX 77226-1258
PH: 800-538-5823
http://www.jetlube.com

Johnson & Johnson, Inc.
Permacel Division
501 George Street
New Brunswick, NJ 08901
201-524-0400

CHART 12 (Sheet 2 of 3)
VENDOR CONTACT INFORMATION

K

Kevlar Special Products
E.I. DuPont de Nemours & Co., (Inc.)
Textile Fibers Department
Centre Road Building
Wilmington, DE 19898
302-999-3156

L

Lehigh - Tenneco Chemicals Co., Inc.
Chestertown, MD 21620
301-778-1991

Loctite Corporation
777 N. Mountain Road
Newington, CT 06111
800-243-8160
In CT 800-842-0225

LPS Laboratories
4647 Hugh Howell Rd.
Tucker, GA 30084
800-241-8334
http://www.lpslabs.com/

LUBRIPLATE Lubricants Co.
129 Lockwood St.
Newark, NJ 07105
PH: 800-733-4755
http://www.lubriplate.com

M

Miller-Stephenson
George Washington Hwy.
Danbury, CT 06810
PH: 203-743-4447
http://www.miller-stephenson.com

Minnesota Mining and MFG
3M Center
St. Paul, MN 55144
612-733-1110

Mirror Bright Polish Co., Inc.
Irvine Industrial Complex
P.O. Box 17177
Irvin, CA 92713
714-557-9200

Mobil Oil Corporation
150 E. 42nd Street
New York, NY 10017
212-883-4242

Morton Inc.
7341 Anacona Ave
Garden Grove, CA 92641
724-373-2837
Fax 724-373-1913

N

Norton Tape Division
Department 6610
Troy, NY 12181
518-273-0100

P

Parker Hannifin Corp.
O-Ring Division
2360 Palumbo Drive
Lexington, KY 40509
PH: 859-269-2351
http://www.parker.com

Parker Seal Company
17325 Euclid Avenue
Cleveland, OH 44112
216-531-3000

Permatex Co., Inc.
P.O. Box 11915
Newington, CT 06111
203-527-5211

PPG Aerospace PRC-DeSoto
11601 United Street
Mojave, California 93501
661-824-4532
818-549-7999
http://corporateportal.ppg.com/na/aerospace/

R

Radiator Specialty Co.
P.O. Box 34689
Charlotte, NC 28234
704-377-6555

Ram Chemicals
201 E. Alondra Blvd.
Gardena, CA 90248
213-321-0710

Rockwell International
600 Grant Street
Pittsburgh, PA 15219
412-565-2000

Royal Lubricants Company
River Road
E. Hanover, NJ 07936
201-887-3100

S

Schnee Moorhead Chemicals, Inc.
PO Box 171305
Irving, TX 75017-1305
(800-878-7876)
http://www.schneemorehead.com/

Shamban W.S. and Co.
1857 Centinela Avenue
Santa Monica, CA 90404
213-397-2195

Shell Oil Company
909 Fannin St STE 700
Houston TX 77010-1016
713-220-6697

Sinclair Refining Co.
600 Fifth Avenue
New York, NY 10020

Socony Mobil Oil Co.
Washington 5, DC 20005

Solar Compounds Corp.
1201 W. Blancke Street
Linden, NJ 07036
201-862-2813

Standard Oil of California
225 Bush Street
San Francisco, CA 94104
415-894-7700

Sun Oil Company of Penna
5 Penn Center Plaza
Philadelphia, PA 19103
215-972-2000

CHART 12 (Sheet 3 of 3)
VENDOR CONTACT INFORMATION

T

Taxacone Company
P.O. Box 10823 TR
Dallas, TX 75208

Texaco, Inc.
2000 Westchester Avenue
White Plains, NY 10650
914-253-4000

Turco Products Inc.
24600 S. Main Street
Box 6200
Carson, CA 90749
213-835-8211

U

U.S. Gulf Corp.
P.O. Box 233
Stoney Brook, NY 11790
212-683-9221

Unelko Corporation
727 E. 110th Street
Chicago, IL 60628

Union Carbide; Plastic Div.
270 Park Avenue
New York, NY 10017
212-551-3763

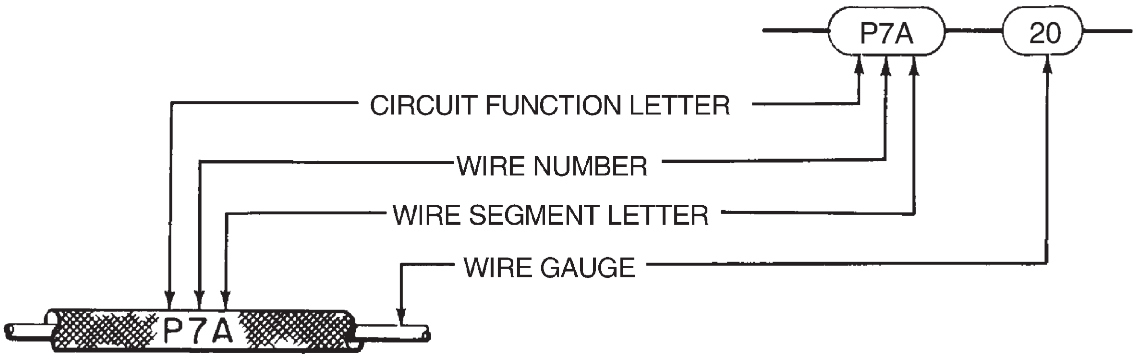
V

Virginia Chemical
3340 W. Norfolk Rd.
Portsmouth, VA 23703
703-484-5000

W

Wooster Products, Inc.
1000 Spruce Street
Wooster, OH 44691
800-321-4936
In OH 216-264-2844

CHART 16 (Sheet 1 of 3)
ELECTRICAL WIRE CODING



(PIR-PPS55006, Rev. AH.)

FUNCTION LETTER (Circa 2010 and Prior)	FUNCTION LETTER (Circa 2010 and Later)	CIRCUIT
AC	H	Air Conditioning
	F	Airspeed Indicator
PF		Alternator Field Control
	F	Altitude Indicator / Altimeter
	L	Anti-collision Lights
	F	Attitude Indicator
A	C	Autopilot
	H	Avionics Cooling
	L	Avionics Lights
	P	Battery
	H	Cabin Heating
	L	Cabin Lights
CP		Cabin Pressurization
	H	Cabin Ventilation
	D	Clock
C		Control Surface
	E	DAU
	C	Electric Trim
	M	ELT Switch
K		Engine Control, Starter
	J	Engine Ignition
E		Engine Instrument
	Q	Engine Priming
	K	Engine Starting
	E	Engine Temperature
	M	Entertainment System
	W	Exceedance Horn
	C	Flap Position/Control
	F	Flight Display/ MFD
FD		Flight Director
F		Flight Instruments
	E	Fuel Flow

CHART 13 (Sheet 2 of 3)
ELECTRICAL WIRE CODING

FUNCTION LETTER (Circa 2010 and Prior)	FUNCTION LETTER (Circa 2010 and Later)	CIRCUIT
	E	Fuel Pressure
	Q	Fuel Pumps
Q	Q	Fuel Qty
	G	Gear Extension and Retraction
	G	Gear Position and Warning
GND		Ground
	T	Hazard Awareness (IHAS, Skywatch, etc.)
H		Heating, Ventilating
	D	Hours Meters
	G	Hydraulic Pump, Power, and Control “
J		Ignition
G		Landing Gear
	L	Landing Lights
L		Lighting
	E	Manifold Pressure
M		Misc. Equipment, Cigar Lighter, Hour Meter
	L	Navigation Lights
	E	Oil Pressure
Q		Oil Quantity
	E	Oil Temperature
OX		Oxygen
	W	Oxygen System
	L	Panel Lights
	H	Pitot Heat
	P	Power Distribution
	P	Power Generation
	P	Power Monitoring
X		Power, AC
P		Power, DC
	P	Power, External
PP		Propeller
	K	Propeller Control
	E	Propeller Overspeed Governor
RZ		Radio Audio
RC		Radio Cooling
RG		Radio Gnd
RP		Radio Power
	R	Radio, ADF
	R	Radio, Comm and Nav
	R	Radio, DME
	R	Radio, Headphone, Microphone and Speakers
	R	Radio, Transponder
	L	Recognition Lights
	W	Stall Detection and Warning
S		Stall Warning

CHART 13 (Sheet 3 of 3)
ELECTRICAL WIRE CODING

FUNCTION LETTER (Circa 2010 and Prior)	FUNCTION LETTER (Circa 2010 and Later)	CIRCUIT
SB		Speed Brake
ST		Stereo
	T	Stormscope
	L	Switch Lights
	E	Tachometer
	L	Taxi Lights
	D	Vacuum System
GB		Vent / Defogger
W		Warning
	H	Windshield Heat

CHART 17 (Sheet 1 of 2)
ELECTRICAL SYMBOLS

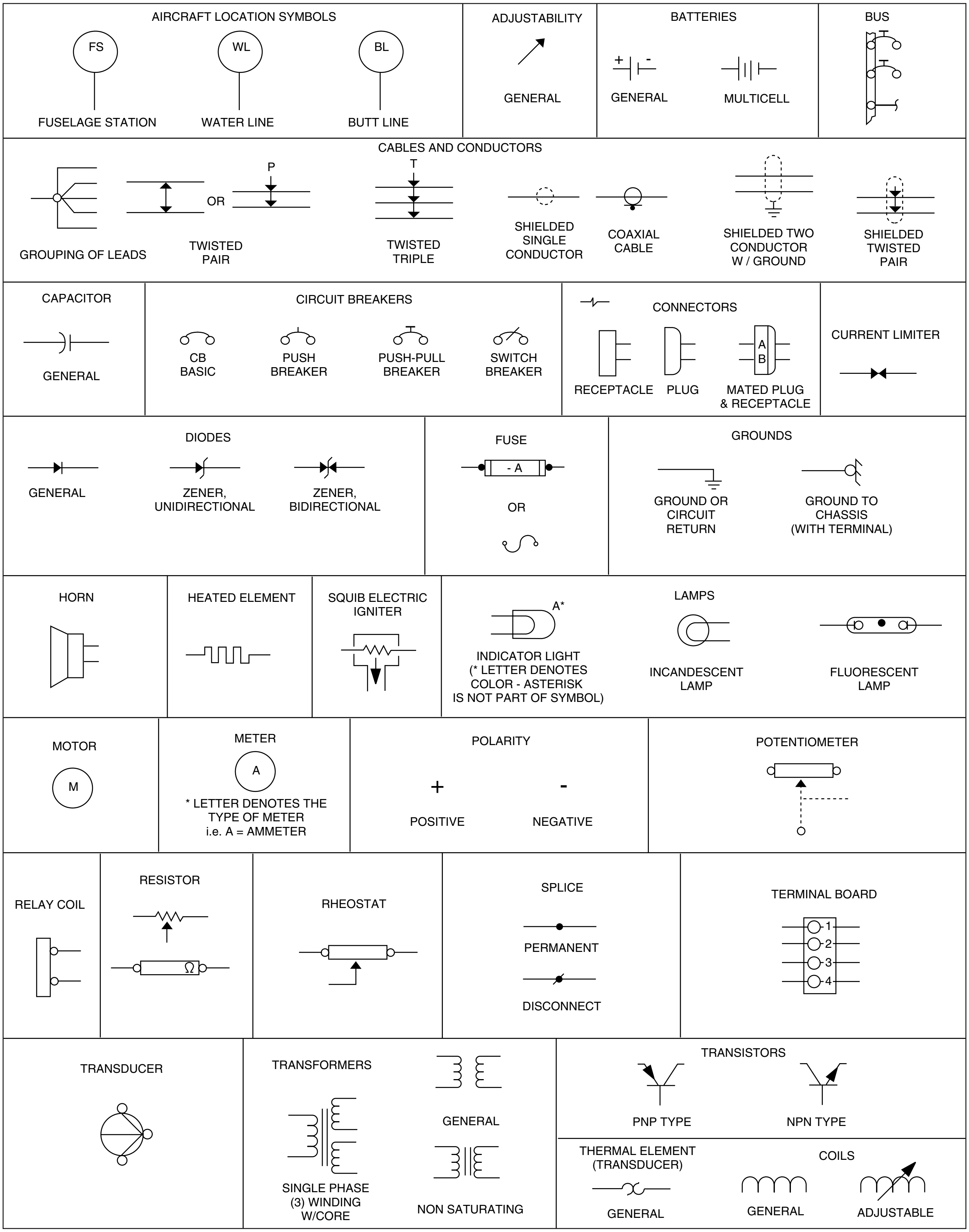


CHART 14 (Sheet 2 of 2)
ELECTRICAL SYMBOLS

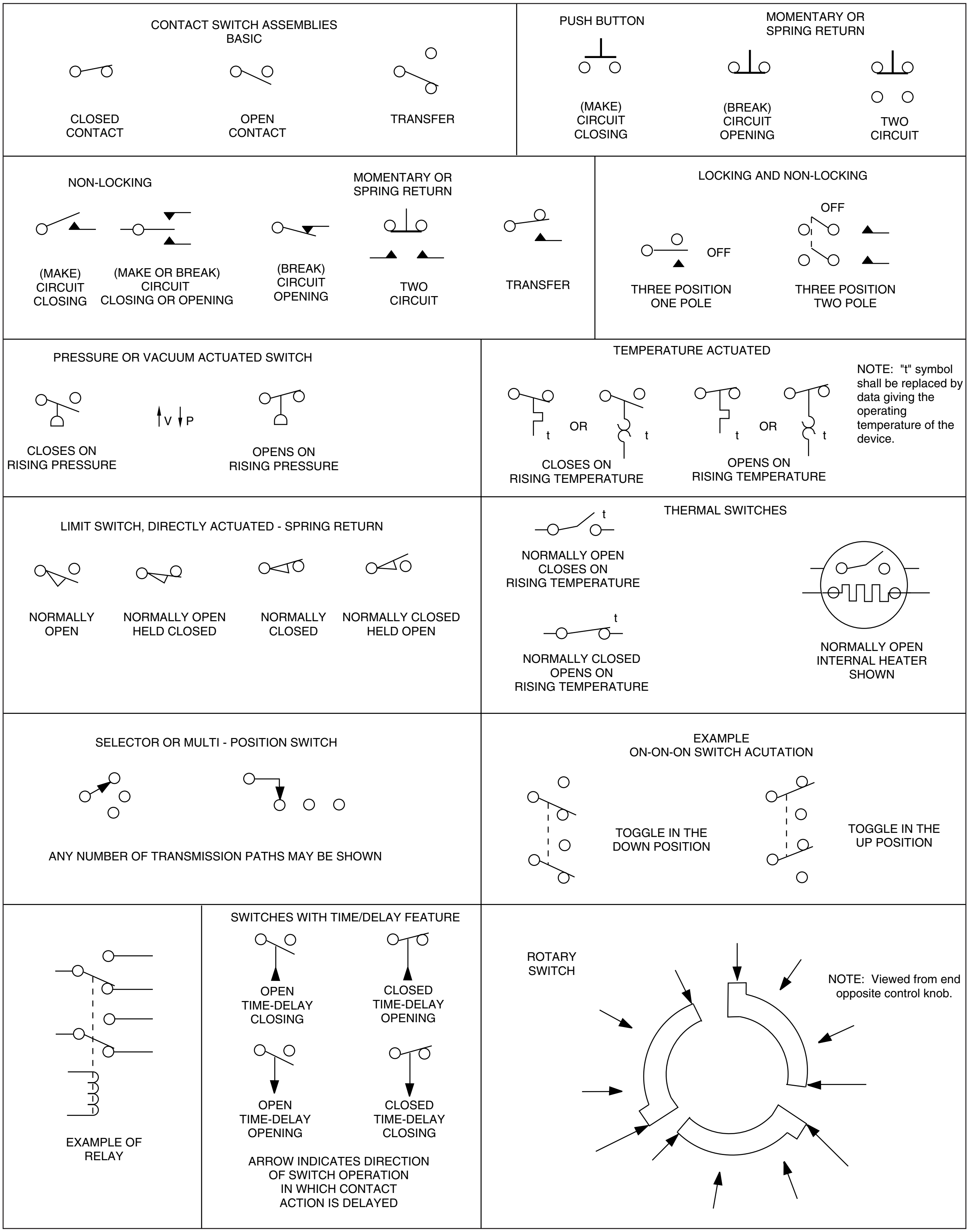


CHART 18
ELECTRICAL / ELECTRONIC COMPONENT REFERENCE DESIGNATION CODES

All electrical / electronic components are assigned a unique reference code which indicates component function and general location within the airplane.

S 301

Class Designation Letter (from list at right)

Component Number (location within airplane indicated by number ranges below:)

100 to 199 -- Electrical - Left Wing and Nacelle
200 to 299 -- Electrical - Right Wing and Nacelle
300 to 399 -- Electrical - Fuselage area (inside cabin)
400 to 499 -- Electrical - Fuselage area (outside cabin)
500 to 599 -- Avionics - Left Wing and Nacelle
600 to 699 -- Avionics - Right Wing and Nacelle
700 to 799 -- Avionics - Fuselage area (inside cabin)
800 to 899 -- Avionics - Fuselage area (outside cabin)

So, the example above, S301, represents either a flasher, switch, or thermostat located in the left fuselage area inside the cabin.

Component	Class Letter
Alternator	G
Annunciator	DS
Antenna	E
Battery	BT
Blower / Fan / Motor	B
Capacitor	C
Circuit Breaker	CB
Clock	M
Compass	M
Contactor	K
Diode	D or CR
Flasher	S
Fuse	F
Generator	G
Heater	HR
Horn	LS
Hour Meter	M
Indicator Lamp	DS
Instrument	M
Jack	J
Light	L
Magneto	G
Miscellaneous Electrical Part	E
Plug	P
Potentiometer	R
Power Supply	PS
Pump	B
Receptacle	J
Rectifier	CR or D
Relay / Solenoid	K
Resistor	R
Sensor	A or MT
Shunt	R
Speaker	LS
Splice	SP
Starter	B
Switch	S
Terminal	E
Terminal Board	TB
Thermocouple	TC
Thermostat	S
Transducer	A or MT
Transformer	T
Transistor	Q
Voltage Regulator	VR
Zener Diode	D

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WIRING DIAGRAM (SCHEMATICS) CROSS-REFERENCE

Each schematic in this manual contains a drawing index number in one of the corners of the schematic, showing the drawing number, revision level, and sheet number (i.e., 105553 A 23) of the Piper engineering drawing the schematic was created from.

Later schematics in this manual were created from harness assembly drawings which typically show only portions of a functional system's wiring on each sheet. In harness assembly drawings, internal cross-reference is to other numbered sheets within that same drawing - typically shown (SH 15) = sheet 15. The following provides sheet locating information by drawing number, sheet number and the Section Number, Figure Number, and Sheet Number where that harness assembly drawing sheet is used in this manual.

To follow a drawing internal cross-reference found in a schematic, note the sheet number you are looking for, note the engineering drawing number from the drawing index number found in the corner of the schematic, compare both to the following information to determine where your cross-reference sheet is used.

Engineering		Maintenance Manual		
Drawing Number	**Sheet Number**	**Section Number**	**Figure Number**	**Sheet Number**
85508 (L)	2	91-24-30	1	3
Electrical Schematic	3	91-74-20	1	3
4636022-4636131	4	91-77-40	1	6
	5	91-31-50	2	2
	6	91-77-40	1	3
	7	91-77-40	1	4
	8	91-31-30	1	3
	9	91-28-40	1	3
	10	91-28-20	1	3
	14	91-31-50	1	4
	14.1	91-31-50	1	5
	17	91-33-40	1	3
	17.1	91-33-40	2	3
	18	91-33-40	4	3
	19	91-33-10	4	3
	20	91-33-10	3	3
	23	91-33-10	1	3
	26	91-30-30	1	3
	26	91-30-30	2	3
	27	91-37-20	1	2
	29	91-21-30	1	3
	33	91-25-10	3	2
	42	91-25-10	4	2
	43	91-77-20	3	2
	45	91-77-40	1	5
	46	91-77-40	1	2

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
89801 (AH)	2	91-24-30	1	2
Electrical Schematic	2.1	91-24-30	2	2
4636001-4636020	3	91-74-20	1	2
	4	91-79-30	1	2
	5	91-77-10	1	2
	6	91-73-30	1	2
	7	91-77-20	1	2
	8	91-31-30	1	2
	9	91-28-40	1	2
	10	91-28-20	1	2
	11	91-27-50	1	2
	12	91-32-60	1	2
	13	91-29-10	1	2
	14.1	91-31-50	1	3
	14.2	91-31-50	1	2
	15	91-32-60	1	3
	16	91-52-70	1	2
	17	91-33-40	1	2
	17.1	91-33-40	2	2
	18	91-33-40	4	2
	19	91-33-10	4	2
	20	91-33-10	3	2
	21	91-33-20	2	2
	22	91-33-20	1	2
	23	91-33-10	1	2
	24	91-33-40	3	2
	25	91-27-30	1	2
	26	91-30-30	1	2
	26	91-30-30	2	2
	28	91-21-50	1	2
	29	91-21-30	1	2
	34	91-30-10	1	2
	30	91-21-40	1	2
	31	91-30-40	1	2
	32	91-30-60	1	2
	34	91-30-10	1	2
	35	91-23-10	1	2
	36	91-33-10	5	2
	37	91-21-20	1	2
	38	91-34-20	1	2

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
89801 (cont.)	39	91-25-10	2	2
	39.1	91-25-10	2	2
	40	91-35-10	1	2
	41	91-25-10	1	2
	42	91-25-10	4	2
	43	91-77-20	2	2
	43.1	91-77-20	3	2
	44	91-33-10	2	2
101238 (E)	2	91-24-30	1	4
Electrical Schematic	2.1	91-24-30	2	3
4633021,	4	91-77-40	2	6
4636132-4636186	6	91-77-40	2	3
	7	91-77-40	2	4
	14	91-31-50	1	6
	14.1	91-31-50	1	7
	18	91-33-40	4	4
	19	91-33-10	4	4
	24	91-33-40	3	3
	28	91-21-50	1	3
	30	91-21-40	1	3
	31	91-30-40	1	3
	32	91-30-60	1	3
	34	91-30-10	1	3
	35	91-23-10	1	3
	39	91-30-10	2	2
	41	91-25-10	4	3
	42	91-77-20	3	2
	43	91-33-10	2	2
	44	91-77-40	2	5
	45	91-77-40	2	2
	46	91-25-10	1	2
101301 (C)	2	91-24-30	1	5
Electrical Schematic	14	91-31-50	1	8
4636187-4636247	14.1	91-31-50	1	9
	28	91-21-50	1	4
	47	91-27-60	1	2

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
104051 (C)	14	91-31-50	1	10
Electrical Schematic	14.1	91-31-50	1	11
4636248-4636313	19	91-33-10	4	5
less 4636299	35	91-23-10	1	4
	46	91-25-10	1	2
104291 (N)	1	91-34-20	3	2
Harness Assy	2	91-34-20	3	3, 4
Meggit Single-side	3	91-34-20	3	5
EFIS option				
104301 (S)	2.1	91-24-30	2	4
Electrical Schematic	14	91-31-50	1	12
4636299,	14.1	91-31-50	1	13
4636314-4636374	14.2	91-31-50	1	14
	17	91-33-40	1	4
	17.1	91-33-40	1	5
	19	91-33-10	4	6
	25	91-27-30	1	3
	26	91-30-30	1	3
	26	91-30-30	2	3
	32	91-30-60	1	4
	37	91-21-20	1	3
	41	91-25-10	4	3
105115 (E rev)	2	91-24-30	1	6
Electrical Schematic	2.1	91-24-30	1	7
Avidyne	3	91-74-20	1	4
	4	91-77-40	3	2
	5	91-77-40	3	3
	6	91-77-40	3	4
	7	91-21-30	1	4
	8	91-28-20	1	4
	9	91-27-50	1	3
	10	91-32-60	1	4
	11	91-29-10	1	3
	12	91-31-50	1	15
	12.1	91-31-50	1	16
	13	91-31-50	2	3
	14	91-33-10	4	7
	14.1	91-33-10	3	4
	14.2	91-33-10	2	3
	15	91-33-20	2	3
	15	91-52-70	1	2

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
105115 (cont.)	16	91-33-20	1	3
	17	91-21-20	1	4
	17	91-33-10	5	3
	18	91-33-10	1	3
	18	91-33-40	3	3
	19	91-33-40	1	6
	20	91-34-20	2	2
	21	91-23-10	1	5
	22	91-27-30	1	4
	23	91-30-30	2	4
	24	91-37-20	1	2
	25	91-21-50	1	5
	26	91-21-40	1	3
	27	91-30-40	1	4
	28	91-30-60	1	5
	29	91-30-10	1	4
	31	91-25-10	4	4
	32	91-27-60	1	3
	33	91-30-30	1	4
105117 (N)	12	91-25-60	2	1
Mirage,	28	91-25-60	2	1
Harness Assy,				
Electrical,				
Avidyne				
105130 (J)	6	91-22-10	1	2
Electrical Schematic	7	91-22-10	1	3
Mirage,	8	91-34-20	4	2
Harness Assy,	9	91-34-20	4	3
Avidyne Entegra	10	91-34-20	4	4
	11	91-34-20	4	5
	12	91-34-20	5	2
	13	91-34-50	1	2
	14	91-34-50	1	3
	15	91-34-50	1	4
	16	91-23-50	1	2
	17	91-23-50	1	3
	18	91-34-40	1	2
	19	91-34-50	2	2
	20	91-34-50	4	2
	21	91-34-40	2	2
	22	91-34-40	3	2

Engineering Drawing Number	Engineering Sheet Number	Maintenance Manual Section Number	Maintenance Manual Figure Number	Maintenance Manual Sheet Number
105130 (cont.)	23	91-34-40	3	3
	24	91-23-15	1	2
	24	91-34-40	4	2
	25	91-34-50	5	2
	26	91-23-50	2	2
	26	91-34-50	6	2
	27	91-34-50	3	2
	28	91-24-30	3	2–3
	29	91-24-30	4	2–3
	30	91-24-30	5	2
105511 (E) Electrical Assy Control Wheel	1	91-22-10	3	2
105551 (AG)	9	91-77-40	4	5
Harness Assy,	10	91-77-40	4	5
Electrical,	14	91-31-50	3	2
G1000	14	91-31-50	3	3
	15	91-30-30	2	1
	30	91-24-30	3	8
105552 (D)	3	91-24-30	1	10
Electrical Schematic	3	91-24-30	1	10
G1000	3	91-24-30	1	10
	4	91-31-50	4	1, 2
	4	91-31-50	4	1, 2
	5	91-24-30	1	11
	5	91-24-30	1	11
	5	91-24-30	1	11
	5	91-24-30	1	11
	6	91-74-20	1	6
	6	91-74-20	1	6
	6	91-74-20	1	6
	7	91-77-40	4	2
	7	91-77-40	4	2
	7	91-77-40	4	2
	7	91-77-40	4	2
	8	91-77-40	4	3
	8	91-77-40	4	3
	8	91-77-40	4	3
	8	91-77-40	4	3
	9	91-28-20	1	5
	9	91-28-20	1	5

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
105552 (cont.)	9	91-28-20	1	5
	9	91-28-20	1	5
	9	91-28-20	1	5
	10	91-31-30	1	5
	10	91-31-30	1	5
	10	91-77-40	4	4
	10	91-77-40	4	4
	11	91-27-50	1	4
	11	91-27-50	1	5
	12	91-32-60	1	6
	12	91-32-60	1	6
	12	91-32-60	1	6
	12	91-32-60	1	6
	12	91-32-60	1	6
	12	91-32-60	1	6
	13	91-29-10	1	4
	14	91-32-60	1	7
	14	91-32-60	1	7
	14	91-32-60	1	7
	14	91-32-60	1	7
	15	91-29-10	2	1
	15	91-29-10	2	1
	15	91-29-10	2	1
	16	91-33-10	4	9
	17	91-33-20	2	5
	17	91-24-30	1	17
	18	91-33-20	1	5
	19	91-34-20	2	4
	19	91-34-20	2	4
	20	91-33-40	1	7
	20	91-33-40	1	7
	20	91-33-40	1	7
	20	91-33-40	1	7
	20	91-33-40	1	7
	21	91-33-10	1	4
	21	91-33-40	3	4
	21	91-33-40	4	6
	21	91-33-40	4	6
	22	91-30-10	1	5
	22	91-30-10	1	5
	23	91-27-30	1	6

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
105552 (cont.)	23	91-27-30	1	6
	24	91-30-30	2	6,7
	24	91-30-30	2	6,7
	24	91-30-30	2	6,7
	25	91-35-20	1	2
	25	91-30-30	1	6
	25	91-30-30	1	6
	25	91-30-30	1	6
	26	91-21-50	1	6
	26	91-21-50	1	6
	27	91-37-20	1	3
	27	91-37-20	1	3
	28	91-30-60	1	6
	28	91-30-60	1	6
	29	91-21-40	1	4
	30	91-30-40	1	5
	31	91-25-60	2	3
	31	91-25-60	2	3
	32	91-27-60	1	4
	32	91-27-60	1	4
	33	91-24-30	6	1
	33	91-24-30	6	1
	33	91-24-30	6	1
	34	91-21-30	1	5
	34	91-21-30	1	6
	34	91-35-10	1	3
	34	91-35-10	1	4
	35	91-24-50	1	3
	36	91-33-10	5	5
	36	91-33-10	5	5
	37	91-34-10	2	2
	37	91-34-40	1	3
	37	91-34-40	1	3
	38	91-33-10	3	7
	38	91-33-10	3	7
	39	91-28-40	1	4
	40	91-33-10	4	10
	40	91-33-10	4	10
	40	91-33-10	4	10
	41	91-24-20	1	5
	42	91-24-20	1	4

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
105553 (H)	5	91-23-50	3	2
Harness Assy,	6	91-23-50	3	3
Avionics,	7	91-23-50	3	4
G1000	8	91-23-50	3	5
	9	91-34-20	8	2
	9	91-34-20	8	3
	10	91-34-20	9	2
	11	91-34-20	10	2
	12	91-34-20	10	3
	13	91-34-10	1	2
	13	91-34-10	1	3
	14	91-34-20	11	2
	15	91-34-20	11	3
	16	91-22-10	2	2
	17	91-22-10	2	3
	18	91-21-20	1	5
	19	91-34-20	12	2
	20	91-21-30	1	7
	21	91-34-50	7	2
	22	91-34-50	7	3
	23	91-34-50	5	4
	24	91-34-50	4	4
	25	91-34-40	6	2
	25	91-34-40	6	3
	26	91-23-15	3	2
	28	91-34-40	2	3
	29	91-22-10	2	4
	31	91-23-50	3	2
	32	91-23-50	3	4
	33	91-24-30	3	7
105555 (AA)	4	91-24-60	1	1
Harness Assy,	11	91-24-60	1	1
Circuit Protection,	5	91-24-60	1	2
G1000	12	91-24-60	1	2
	15	91-24-60	1	2
	7	91-24-60	1	3
	15	91-24-60	1	3
	9	91-24-60	1	4

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
	15	91-24-60	1	4
105801 (J)	4	91-22-10	1	4
Matrix,	5	91-22-10	1	5
Harness Assy,	6	91-34-20	6	2
Avionics,	7	91-34-20	6	3
Avidyne Entegra	8	91-34-20	6	4
	8	91-34-50	6	3
	9	91-23-15	2	2
	9	91-34-20	7	2
	10	91-34-50	1	5
	11	91-34-50	1	6
	12	91-34-50	1	7
	13	91-23-50	1	4
	14	91-23-50	1	5
	15	91-34-50	2	3
	16	91-34-40	5	2
	17	91-34-40	7	2
	18	91-34-50	4	3
	19	91-34-50	5	3
	20	91-22-10	1	6
	21	91-24-30	3	4–5
	22	91-24-30	4	4 & 6
105803 (N)	25	91-33-10	3	6
Matrix,	26	91-24-30	3	6
Harness Assy,	26	91-24-30	4	5
Electrical	26	91-24-30	5	3
105805 (C)	3	91-24-30	1	8
Matrix,	3	91-24-30	1	8
Electrical,	3	91-24-30	1	8
Schematic,	3	91-24-30	1	8
Avidyne Entegra	4	91-24-20	1	2
	4	91-24-20	1	2
	4	91-24-30	1	9
	4	91-24-30	1	9
	4	91-24-30	1	9
	4	91-24-30	1	9
	5	91-74-20	1	5
	5	91-74-20	1	5

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
105805 (cont.)	5	91-74-20	1	5
	6	91-77-40	3	5
	6	91-77-40	3	5
	6	91-77-40	3	5
	6	91-77-40	3	5
	7	91-77-40	3	6
	7	91-77-40	3	6
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	8	91-77-40	3	7
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	9	91-31-30	1	4
	9	91-31-30	1	4
	9	91-31-30	1	4
	10	91-28-20	1	4
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	10	91-28-20	1	4
	10	91-28-20	1	4
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	11	91-27-50	1	3
	12	91-32-60	1	5
	12	91-32-60	1	5
	12	91-32-60	1	5
	12	91-32-60	1	5
	13	91-29-10	1	3
	14	91-31-50	1	17
	14	91-31-50	1	17
	15	91-31-50	1	18
	15	91-31-50	1	18
	15	91-31-50	1	18
	15	91-31-50	1	18
	15	91-31-50	1	18
	15	91-31-50	1	18
	16	91-31-50	2	3
	17	91-33-10	4	8
	17	91-33-10	4	8
	18	91-33-10	3	5

Engineering Drawing Number	Sheet Number	Maintenance Manual Section Number	Figure Number	Sheet Number
105805 (cont.)	18	91-33-10	3	5
	19	91-33-20	2	4
	19	91-33-20	2	4
	20	91-33-20	1	4
	21	91-21-20	1	4
	21	91-33-10	5	4
	22	91-33-10	1	3
	22	91-33-40	3	3
	22	91-33-40	4	5
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	27	91-30-30	2	5
	27	91-30-30	2	5
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	34	91-25-60	2	2
	34	91-25-60	2	2
	35	91-27-60	1	3
	36	91-30-30	1	5
	36	91-30-30	1	5

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107951 (P)	10	91-77-40	4	9
Harness Assy,	11	91-77-40	4	9
Electrical,	26	91-24-30	3	9
G1000	27	91-25-60	3	
	27	91-35-10	1	6
107963 (E)	2	91-34-40	8	2
Harnes Assy,				
GTS 825				
107975 (NEW)	5	91-24-30	1	12
Electrical,	5	91-24-30	1	12
Schematic,	5	91-24-30	1	12
Avionics,	5	91-24-30	1	12
463663, 4636652	9	91-24-30	1	13
and Up	9	91-24-30	1	14
	9	91-24-30	1	14
	6	91-24-30	2	5
	6	91-24-30	2	5
	6	91-24-30	2	5
	7	91-24-30	6	2
	7	91-24-30	6	2
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	12	91-77-40	4	7
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	18	91-28-40	1	7
	19	91-32-60	1	8
	19	91-32-60	1	8
	19	91-32-60	1	8
	19	91-32-60	1	8
	19	91-32-60	1	8

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107975 (cont.)	19	91-32-60	1	8
	20	91-29-10	2	2
	20	91-29-10	2	2
	21	91-29-10	1	5
	21	91-29-10	1	5
	21	91-29-10	1	5
	22	91-33-10	5	6
	23	91-33-10	3	7
	24	91-33-10	4	11
	25	91-33-20	2	6
	25	91-33-20	2	6
	26	91-33-20	1	5
	27	91-33-10	1	4
	27	91-33-40	4	7
	28	91-33-40	1	9
	28	91-33-40	1	9
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	30	91-27-50	1	6
	31	91-27-30	1	7
	32	91-30-30	1	7
	32	91-30-30	1	7
	32	91-30-30	1	9
	32	91-30-30	1	9
	33	91-30-30	1	9
	34	91-30-30	2	8
	35	91-21-50	1	7
	35	91-21-50	1	7
	36	91-21-20	1	6
	36	91-21-20	1	6
	36	91-21-20	1	6
	37	91-21-40	1	5
	38	91-31-30	1	6
	39	91-25-60	2	4
	40	91-21-30	1	8
	41	91-30-10	1	6
	42	91-33-40	3	4
	43	91-37-20	1	4
	44	91-30-60	1	7
	45	91-30-40	1	5
	47	91-27-60	1	4
	49	91-24-20	1	6

Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107975 (cont.)	50	91-24-60	5	1
	50	91-24-60	5	2
	51	91-31-20	1	
	52	91-34-20	2	5
	53	91-23-50	4	2
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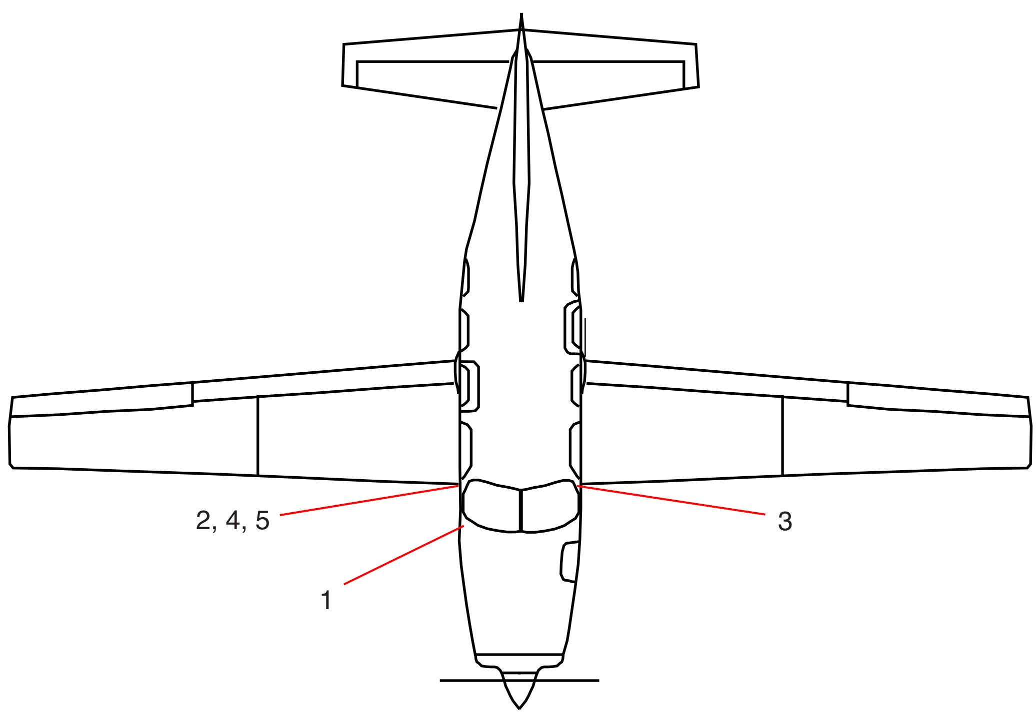
	57	91-34-20	8	4
	58	91-34-20	9	3
	59	91-34-20	10	4
	60	91-34-20	10	5
	61	91-34-10	1	4
	62	91-34-10	1	5
	63	91-34-20	11	4
	64	91-34-20	11	5
	65	91-22-10	2	5
	66	91-22-10	2	6
	67	91-22-10	2	7
	68	91-21-30	1	9
	68	91-21-30	1	9
	70	91-34-50	7	4
	71	91-34-50	7	5
	72	91-34-50	7	6
	73	91-34-40	8	
	74	91-23-15	3	3
	75	91-23-15	4	
	76	91-34-40	1	4
	77	91-34-40	2	4
	78	91-34-50	8	
	79	91-34-50	4	5
	80	91-24-60	3	1
	81	91-24-60	3	2
	82	91-24-60	4	
107991 (NEW)	4	91-34-20	8	5
Harness Assy,	5	91-34-20	9	4
Avionics,	6	91-34-20	10	6
G1000 NXi	7	91-34-20	10	7
	8	91-34-10	1	6
	9	91-34-20	11	6
	10	91-34-20	11	7
	11	91-32-60	1	9

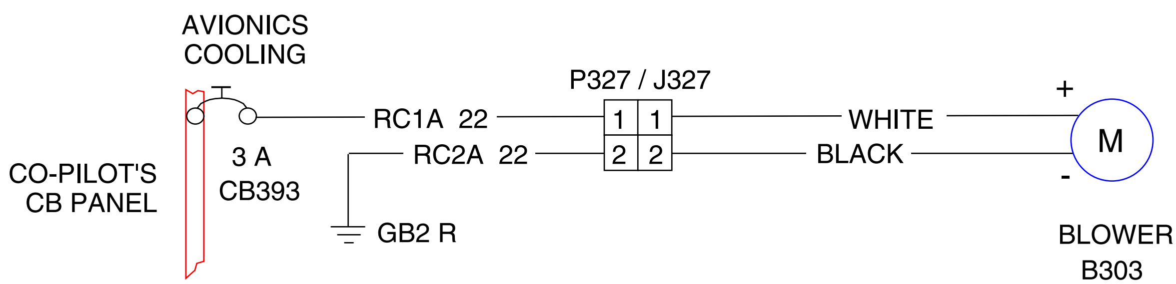
Engineering		Maintenance Manual		
Drawing Number	Sheet Number	Section Number	Figure Number	Sheet Number
107991 (cont.)	12	91-22-10	2	8
	13	91-22-10	2	9
	14	91-21-20	1	7
	14	91-33-10	5	7
	15	91-34-50	9	2
	16	91-34-50	9	3
	17	91-34-50	9	4
	18	91-34-50	9	5
	19	91-34-40	2	5
	20	91-21-30	1	10
	21	91-22-10	2	10
	21	91-34-20	10	8
	22	91-24-30	3	10

Item #	Designation	Description
1	B303	Blower
2	CB393, CB391, CB23*	Circuit Breaker - Avionics Cooling (3 Amp)
3	CB358**	Circuit Breaker - PFD 1 Fan (3 Amp)
4	CB376**	Circuit Breaker - Av/MFD Fan (3 Amp)
5	CB390**	Circuit Breaker - PFD 2 Fan (3 Amp)

* 3 Amp Avionics Cooling Circuit Breaker (4692001 and up - Entegra)
** 3 Amp Circuit Breaker (with G1000 installation)

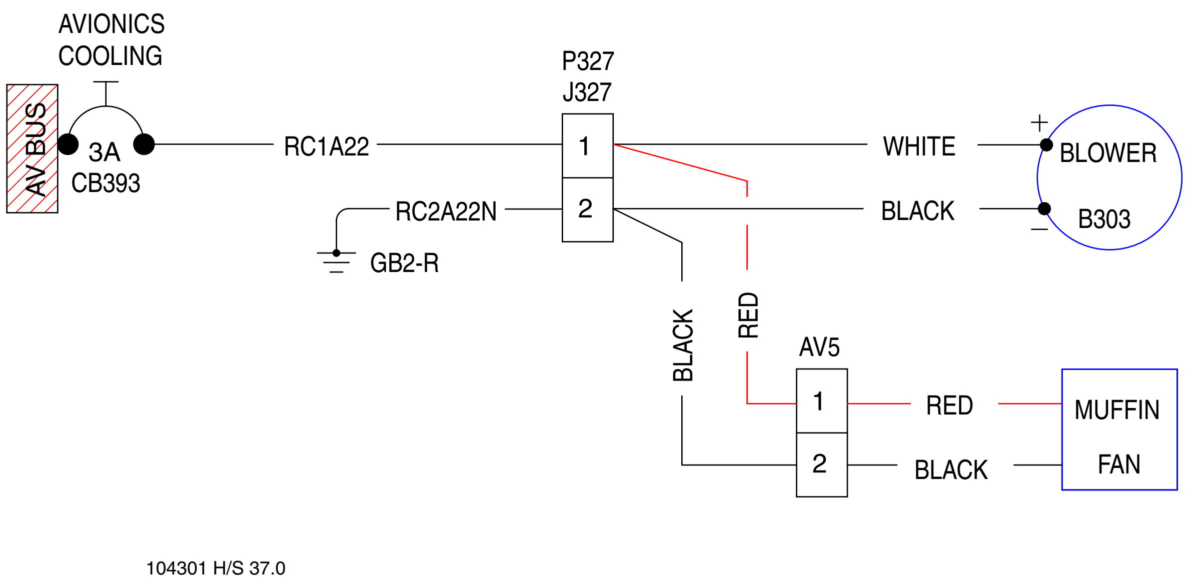


Avionics Cooling
Figure 1 (Sheet 1 of 7)

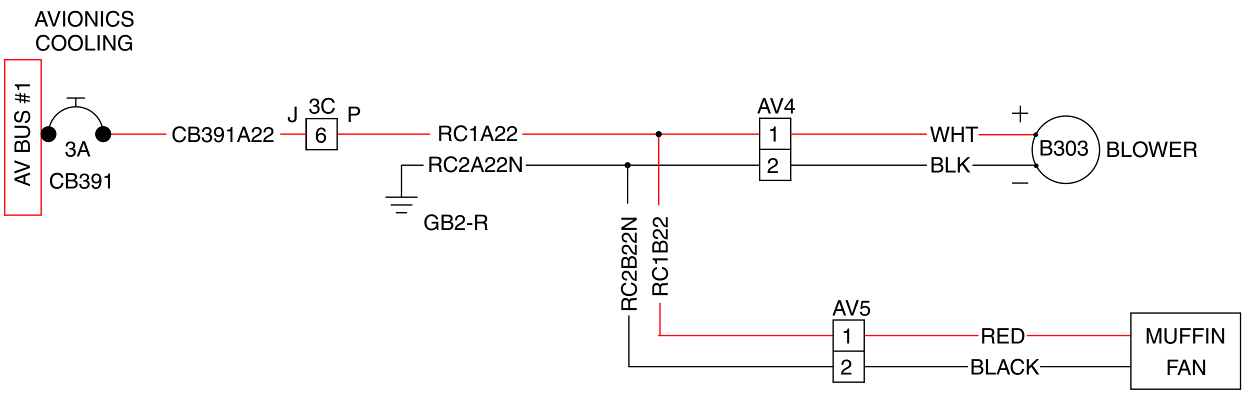


89801 AH 37.0
85508 L
101238 E
101301 C
104051 C

Avionics Cooling
Figure 1 (Sheet 2 of 7)

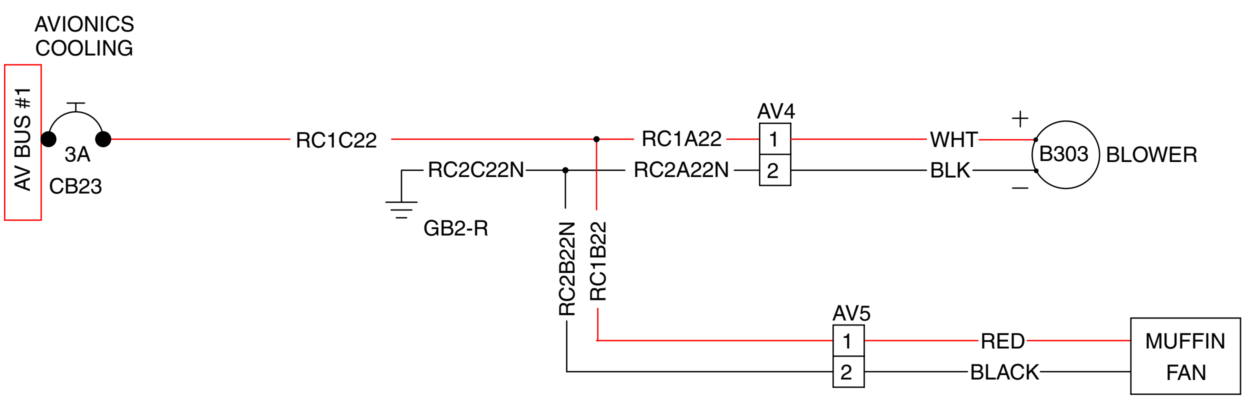


Avionics Cooling
Figure 1 (Sheet 3 of 7)



S/N'S 4636375 AND UP

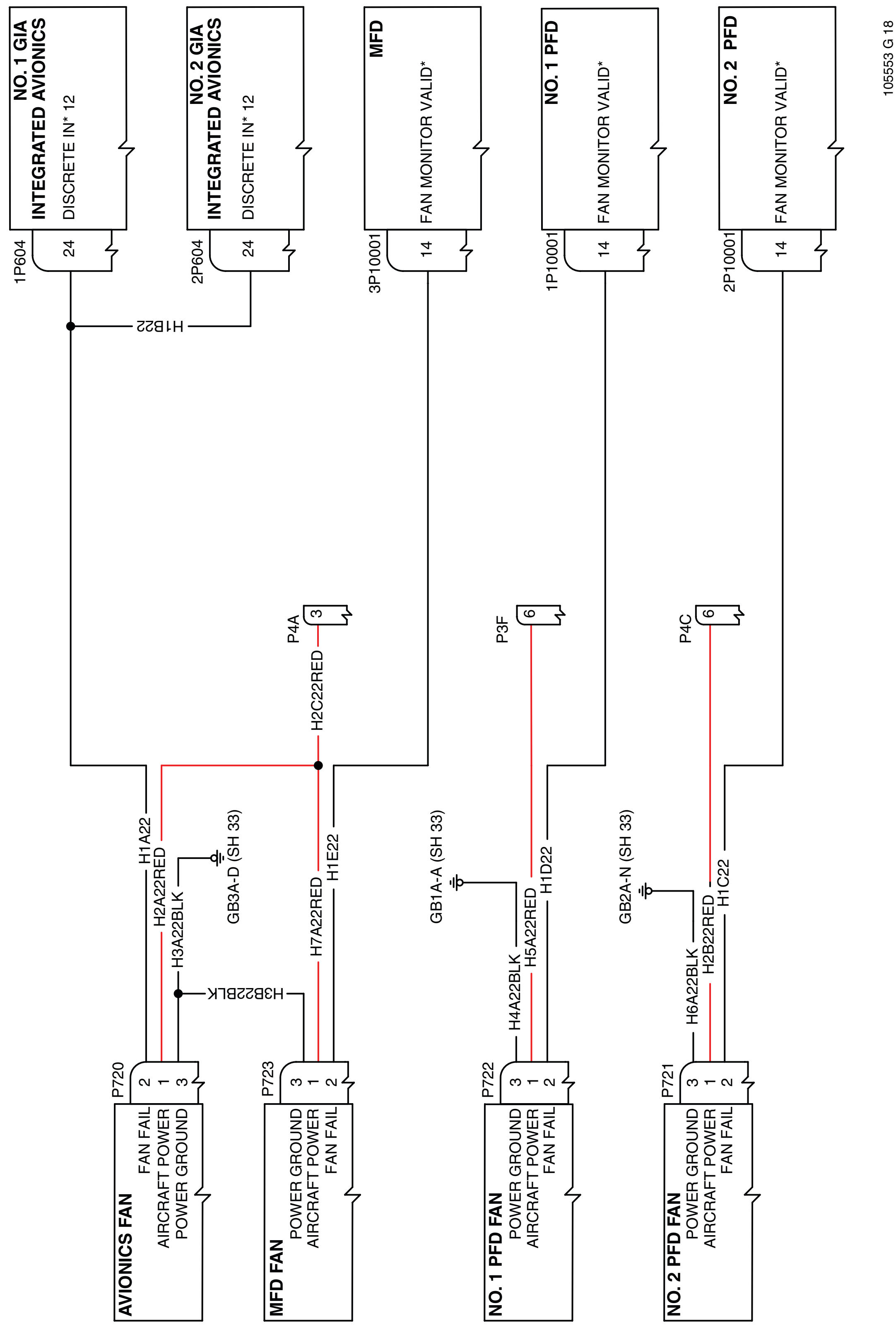
105115 A 17.0



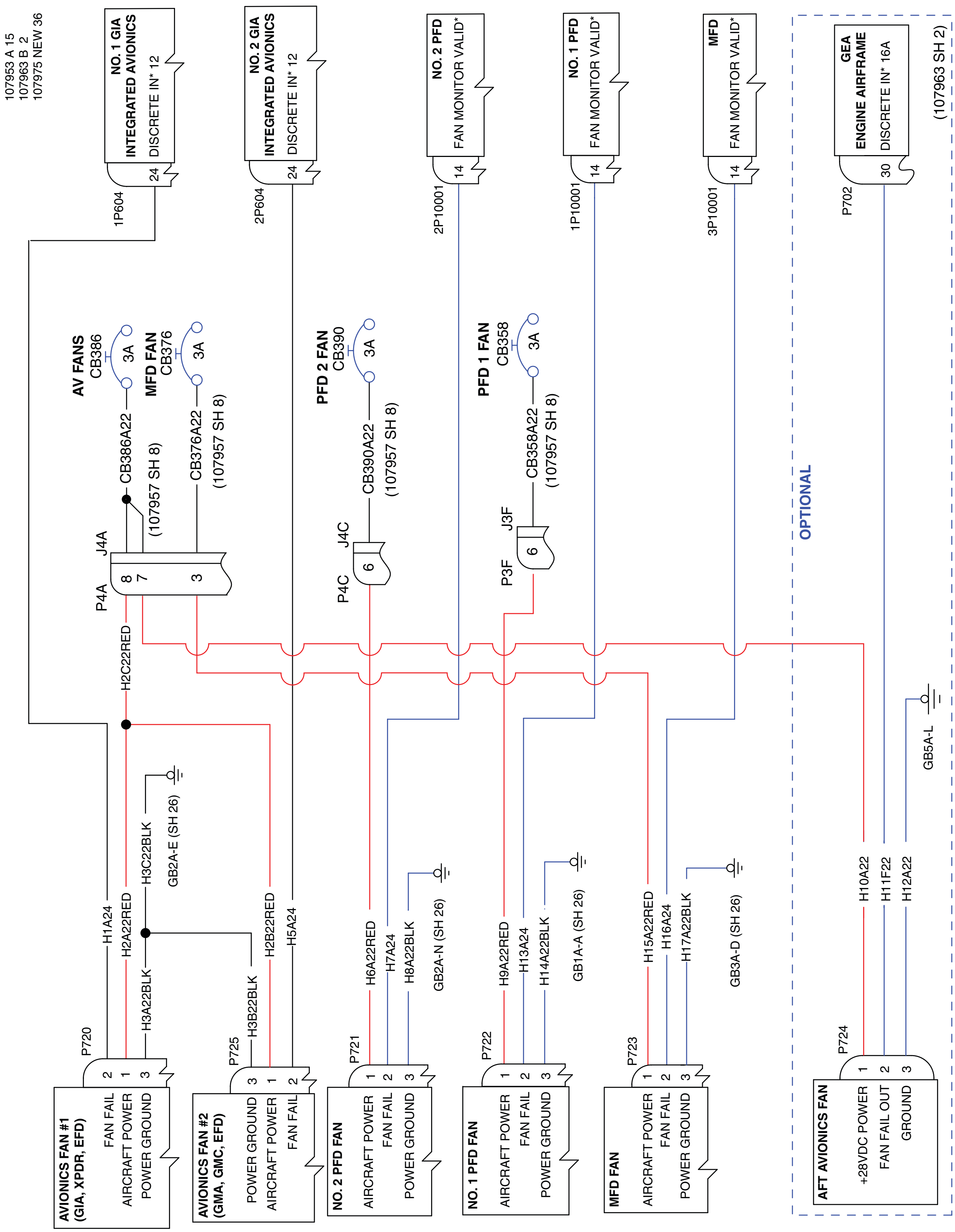
S/N'S 4692001 AND UP

105805 New 21.0

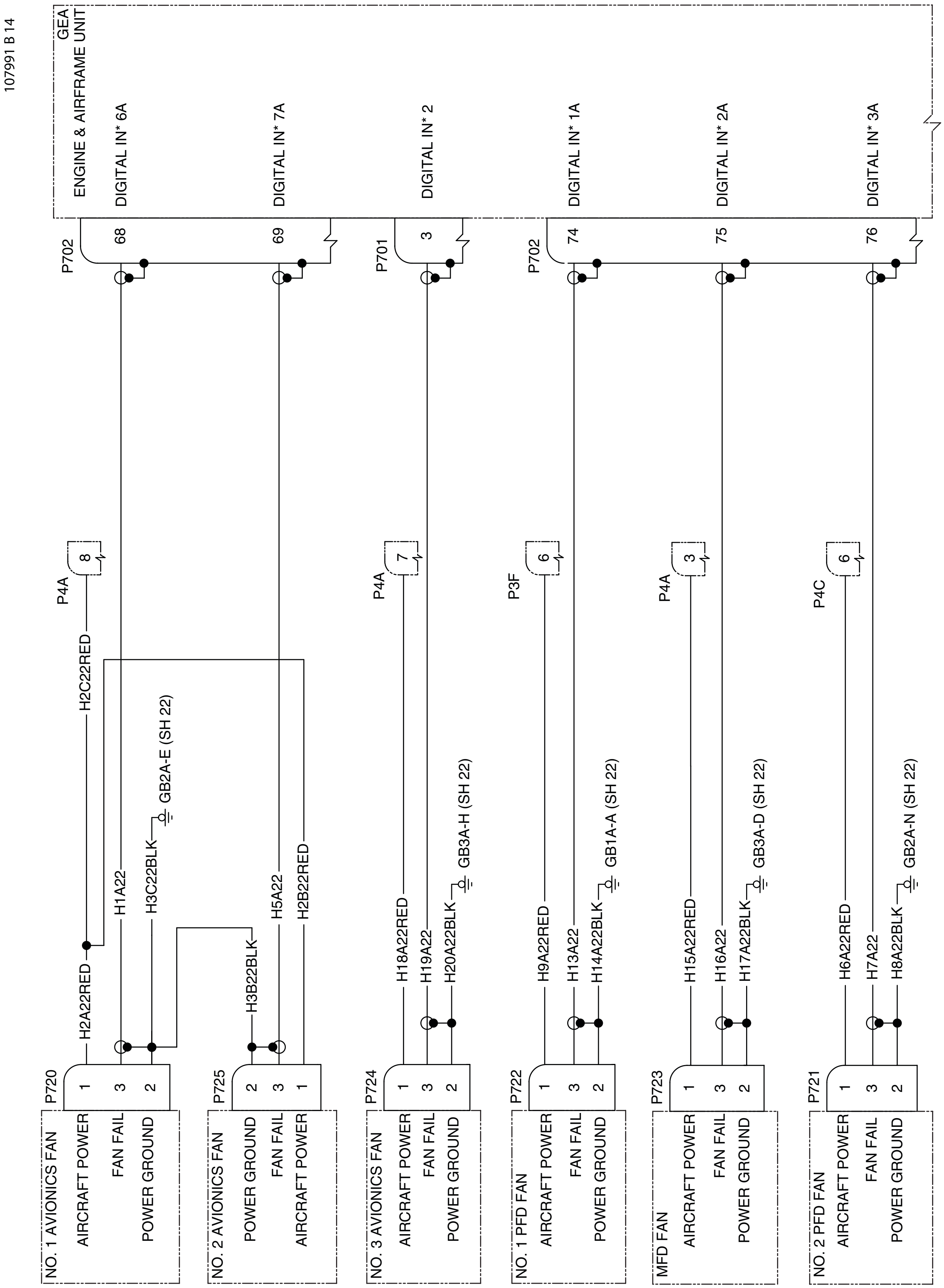
Avionics Cooling
Figure 1 (Sheet 4 of 7)



Avionics Cooling
Figure 1 (Sheet 5 of 7)



Avionics Cooling
Figure 1 (Sheet 6 of 7)



Avionics Cooling
Figure 1 (Sheet 7 of 7)

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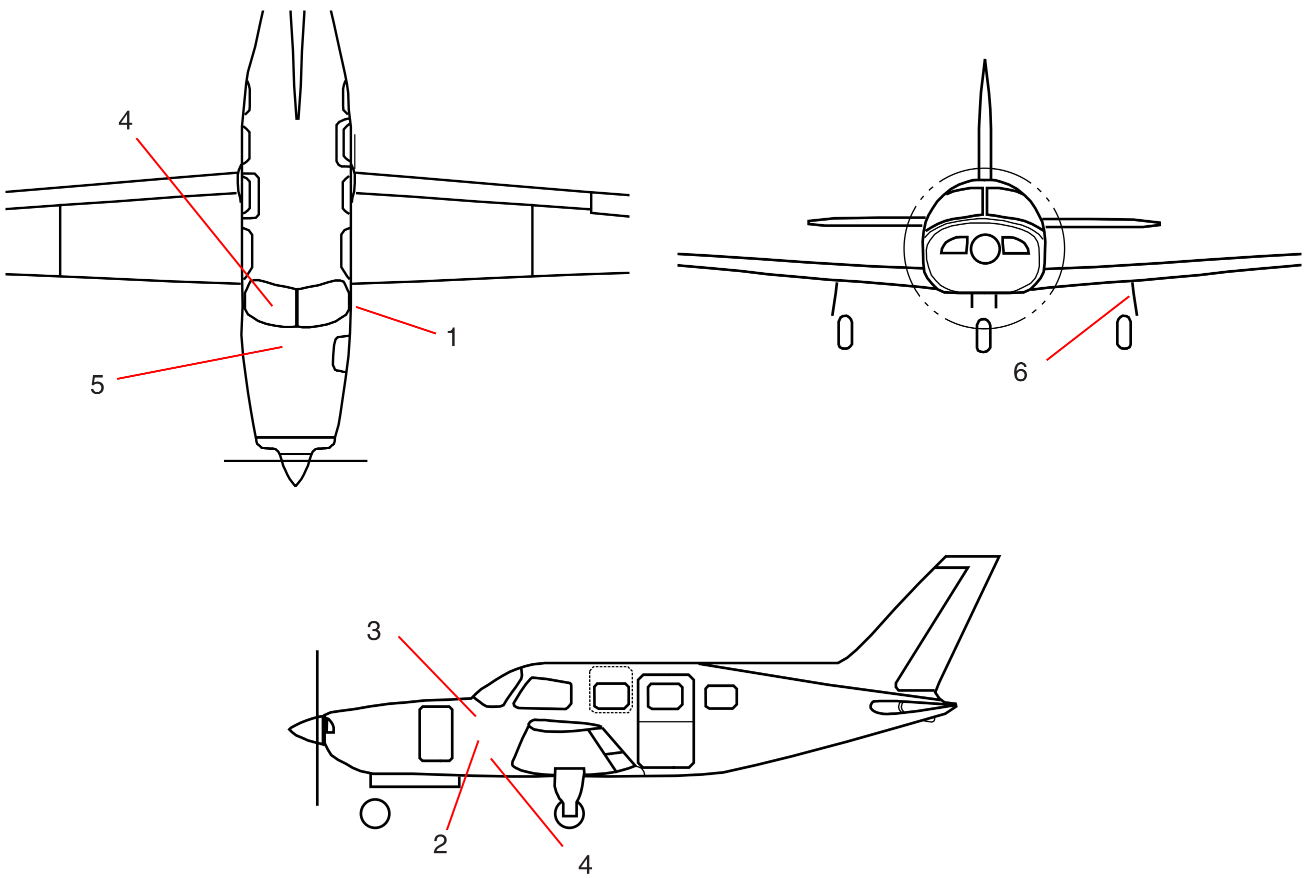
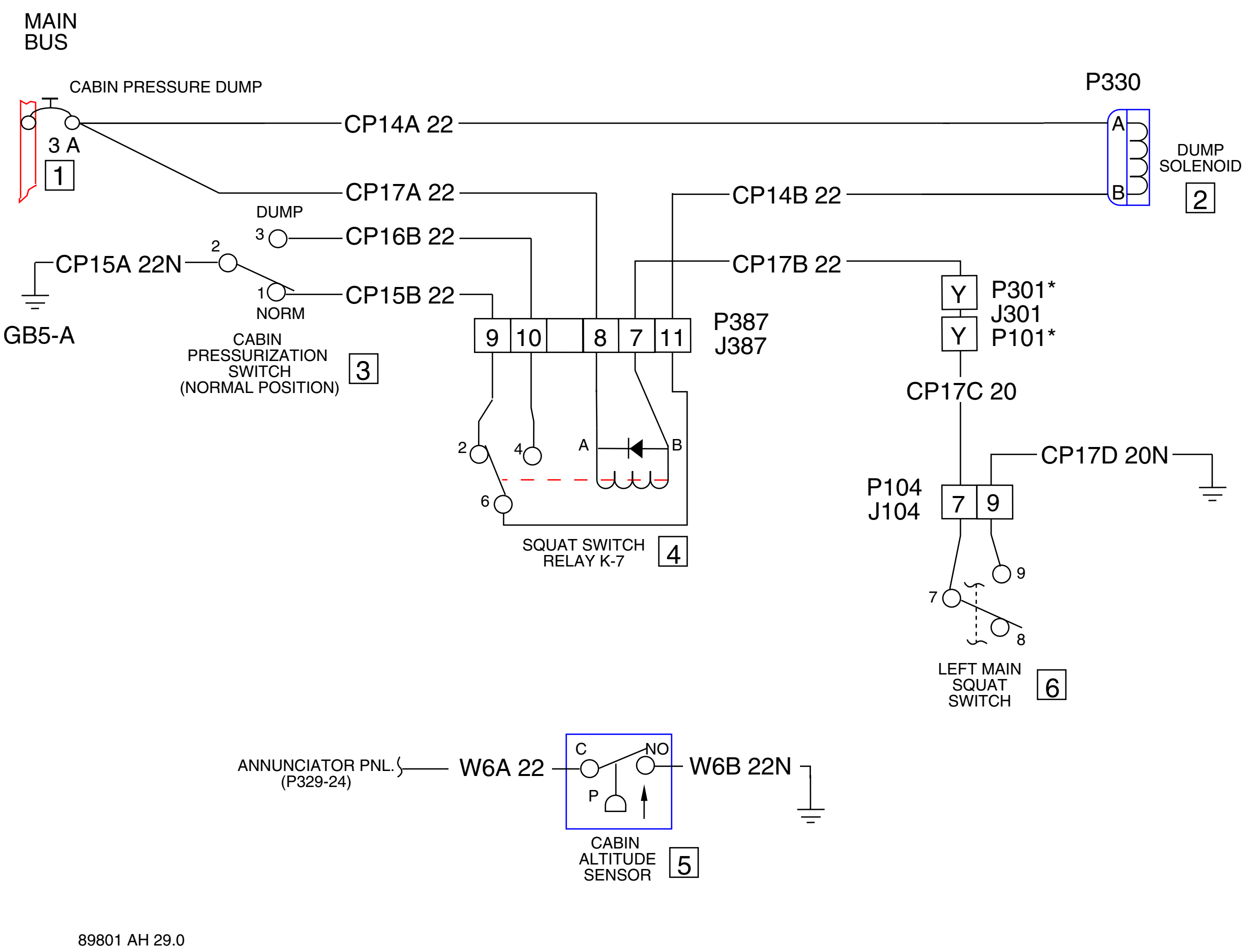
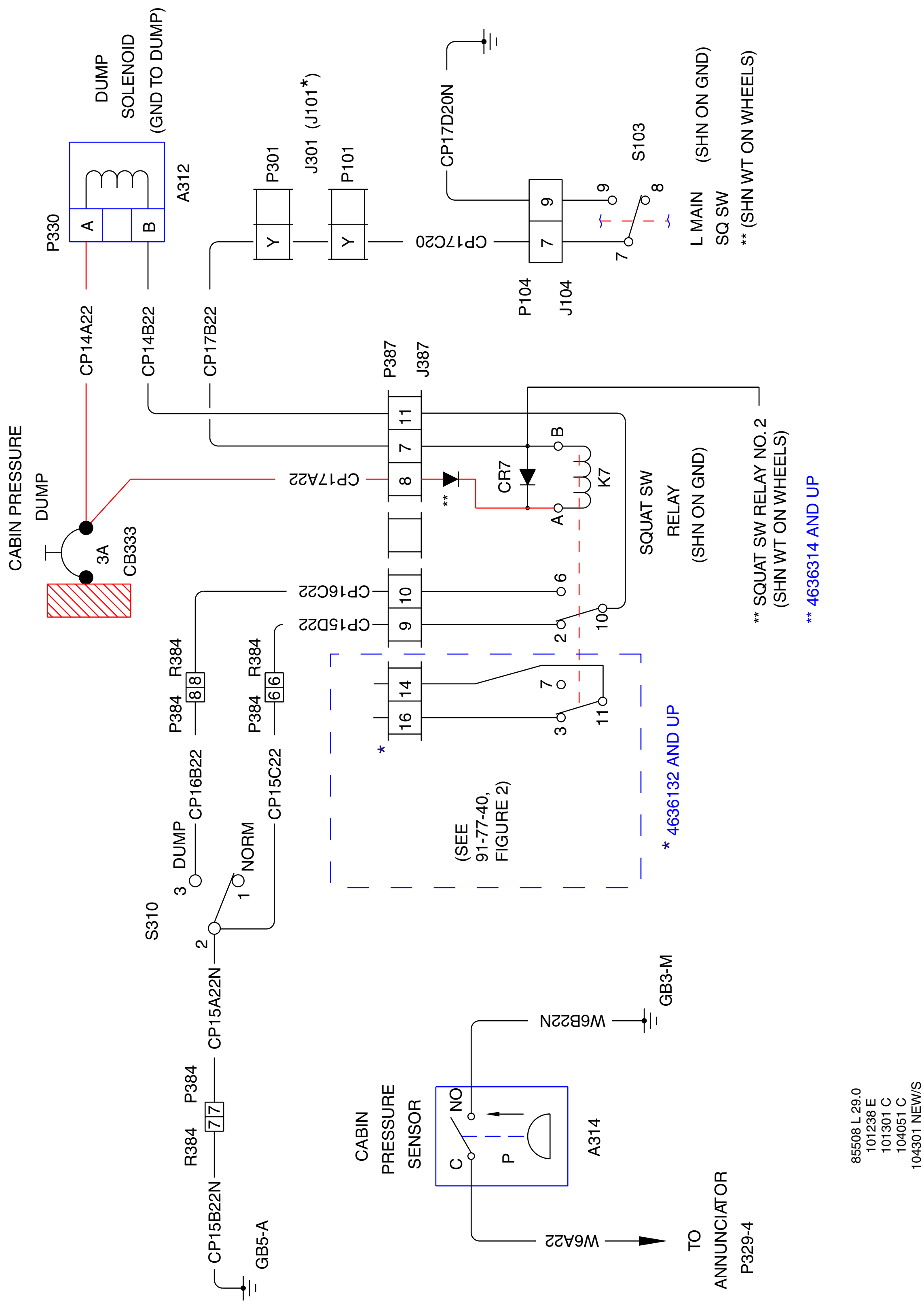
Item #	Designation	Description
1	CB333	Circuit Breaker - Cabin Press. Dump (3 Amp)
2	A312	Cabin Pressurization Dump Solenoid
3	S310	Cabin Pressurization Dump Switch
4	K7	Squat Switch Relay
5	A314	Cabin Pressure Sensor
6	S103	Squat Switch Assembly



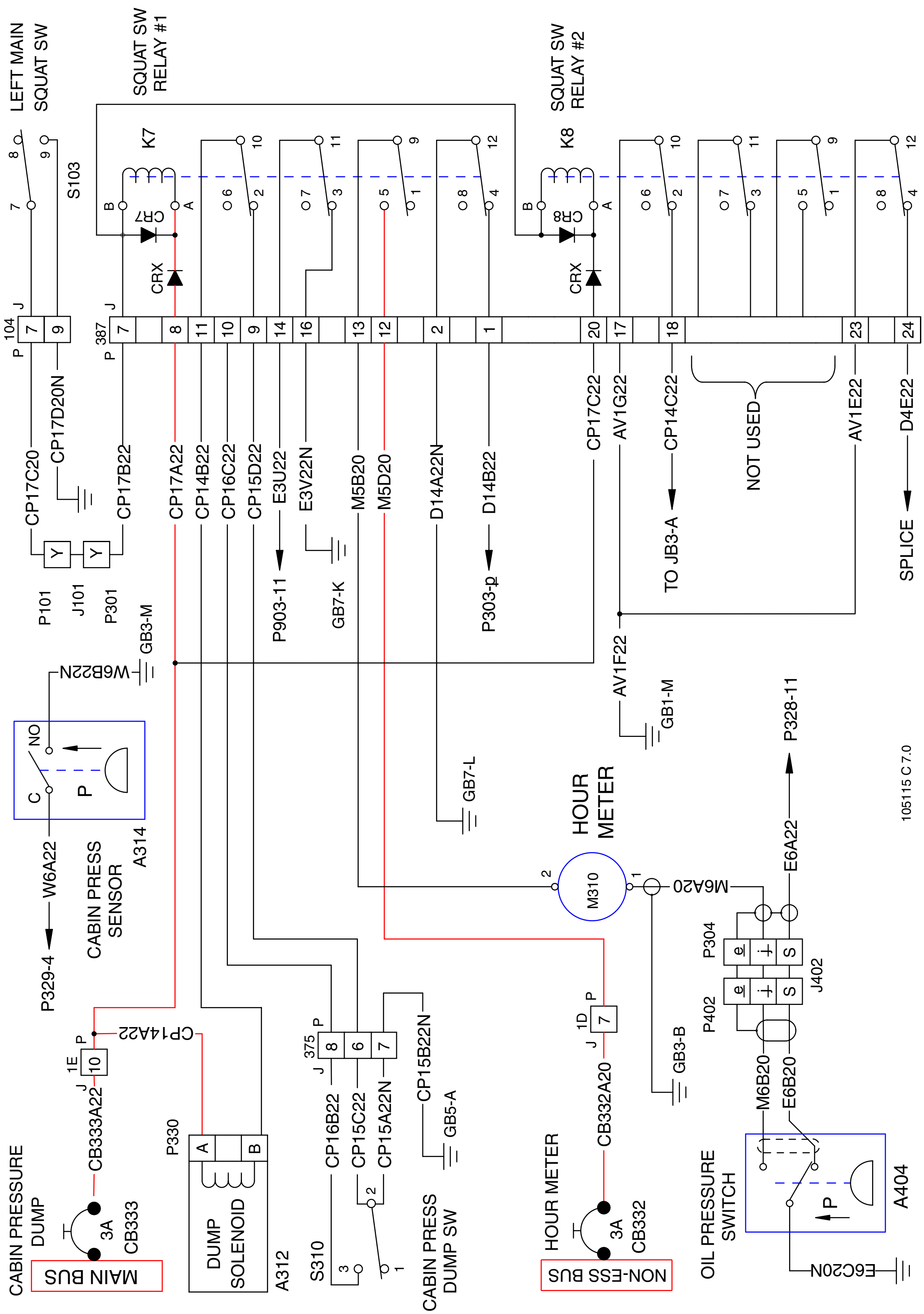
Cabin Pressurization
Figure 1 (Sheet 1 of 10)



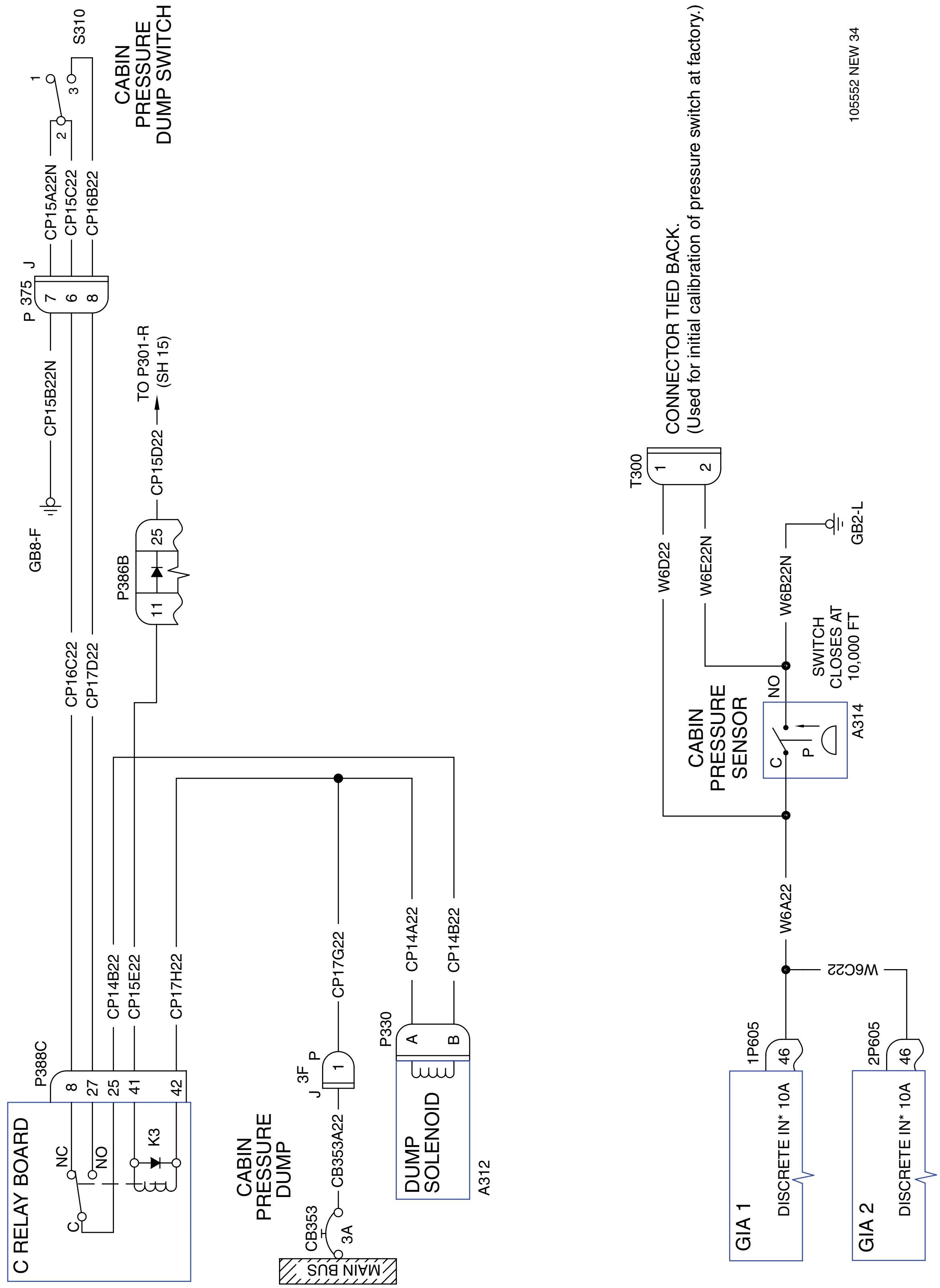
Cabin Pressurization
Figure 1 (Sheet 2 of 10)



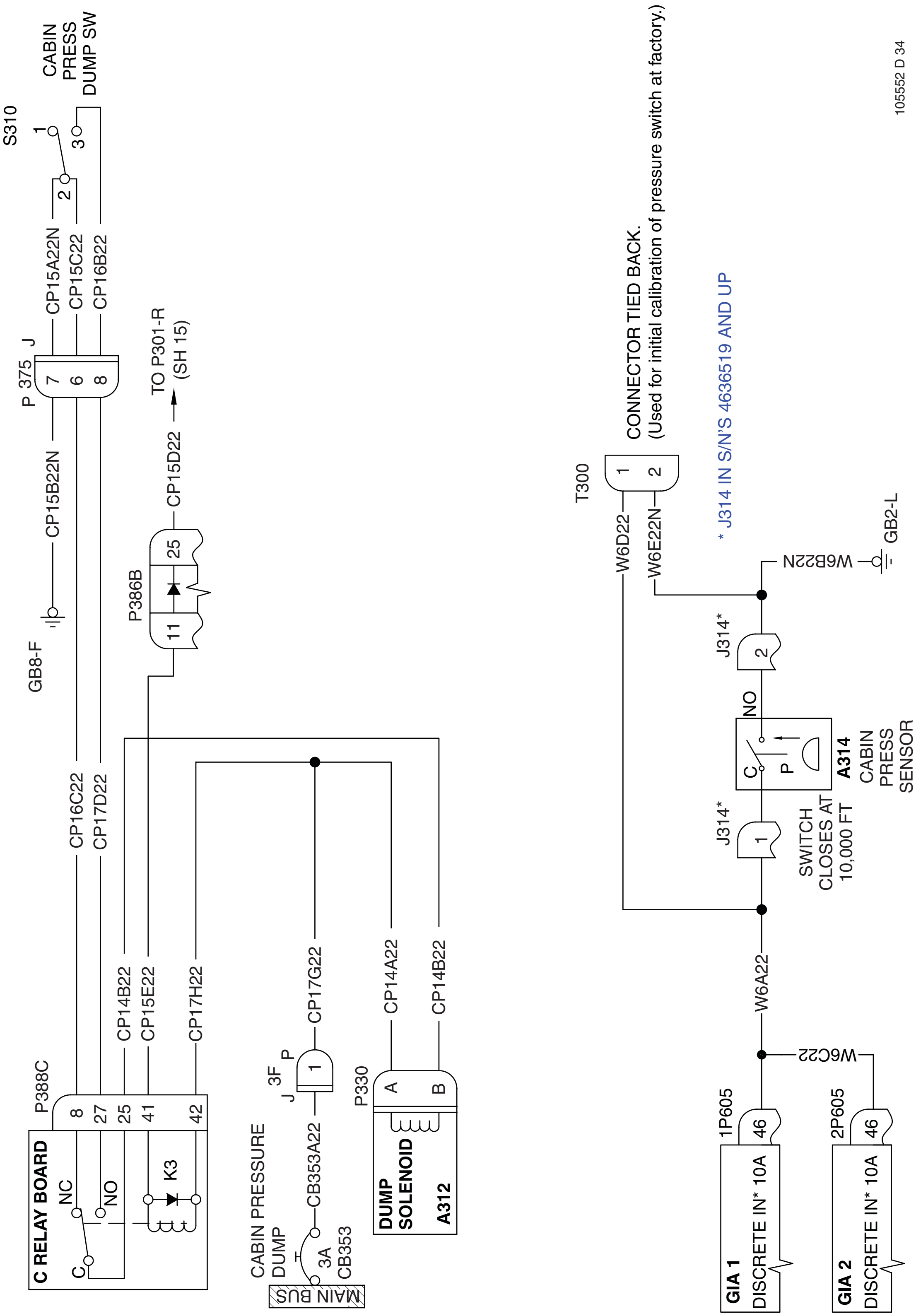
Cabin Pressurization
Figure 1 (Sheet 3 of 10)



Cabin Pressurization
Figure 1 (Sheet 4 of 10)

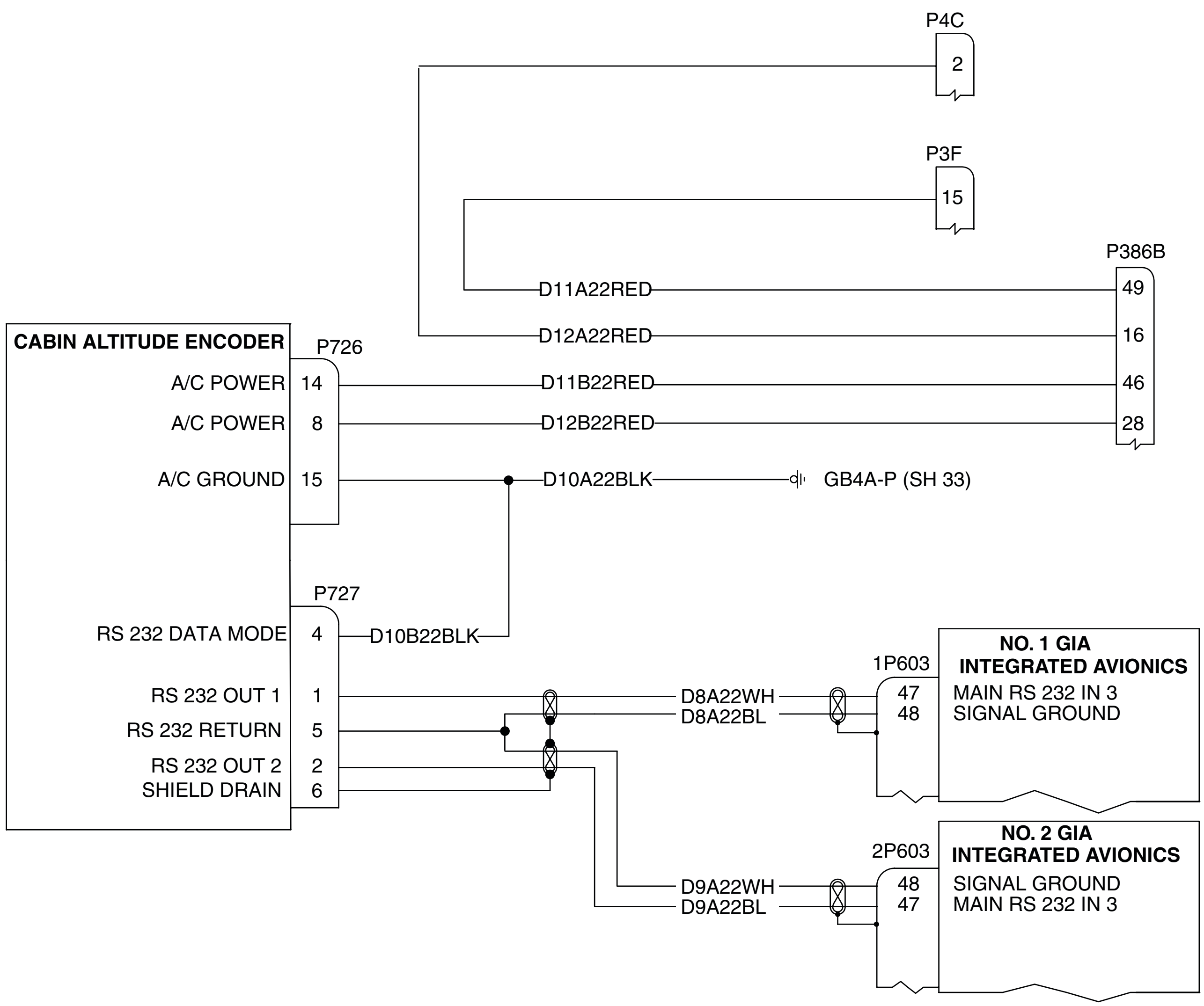


Cabin Pressurization
Figure 1 (Sheet 5 of 10)

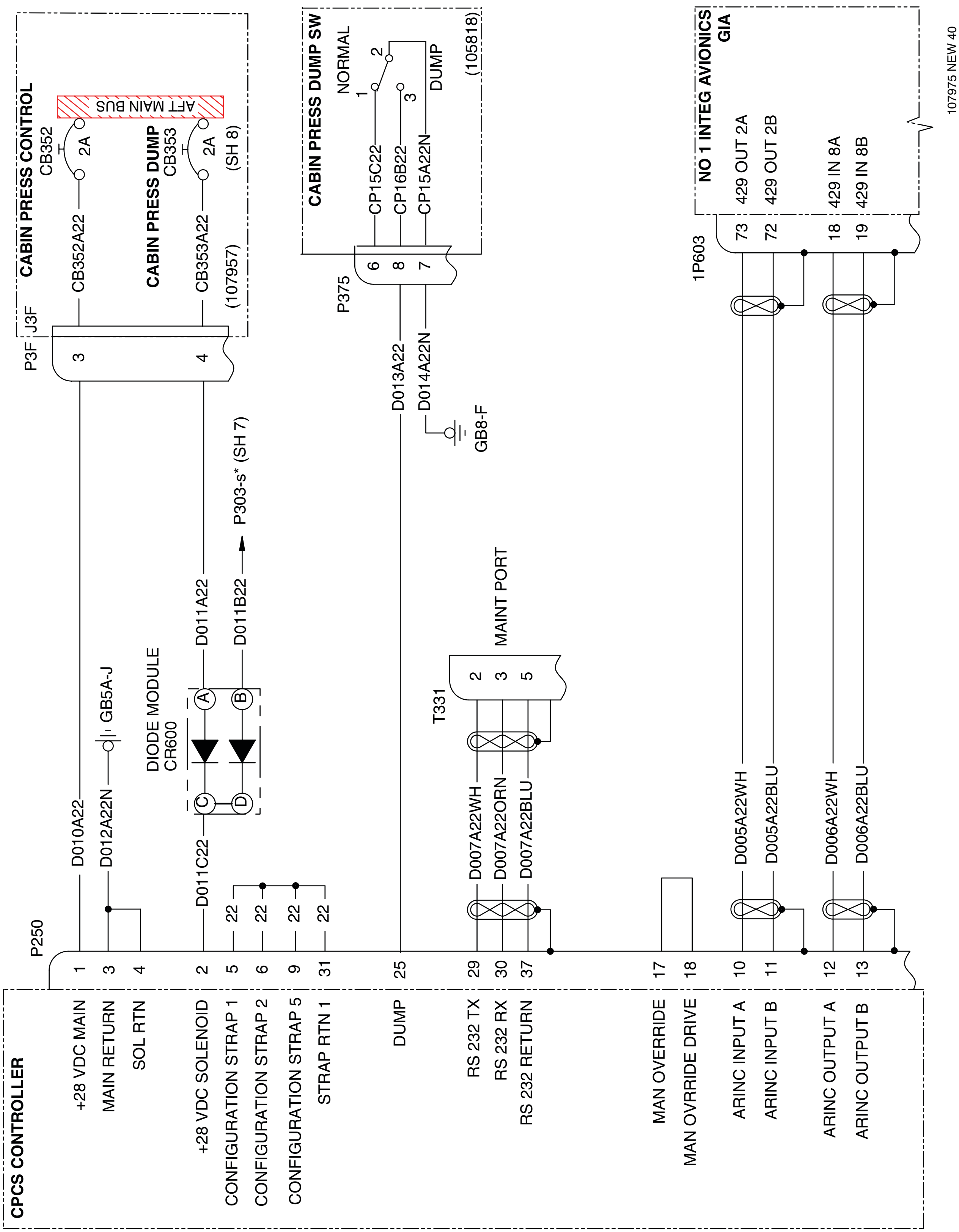


Cabin Pressurization
Figure 1 (Sheet 6 of 10)

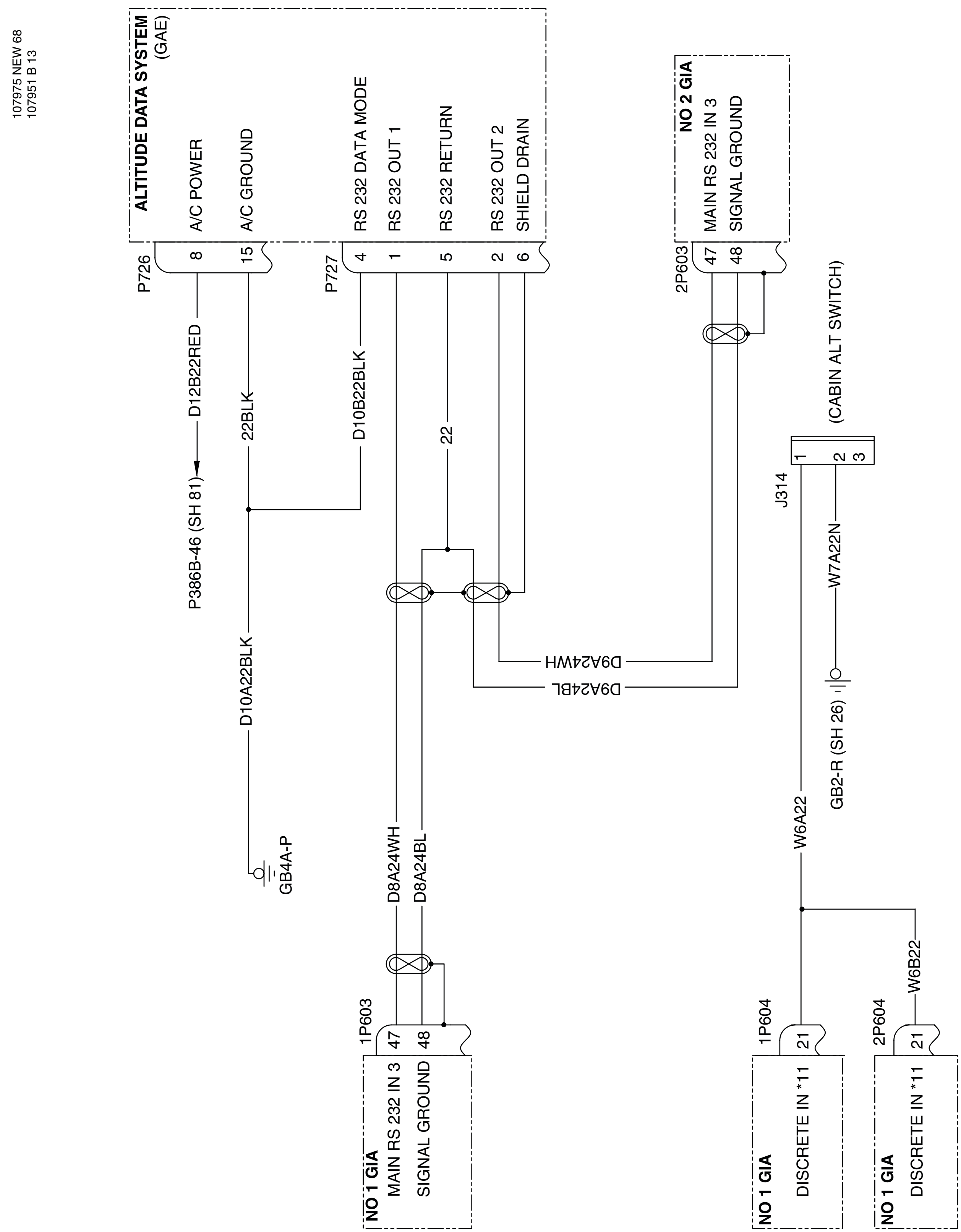
105553 G 20



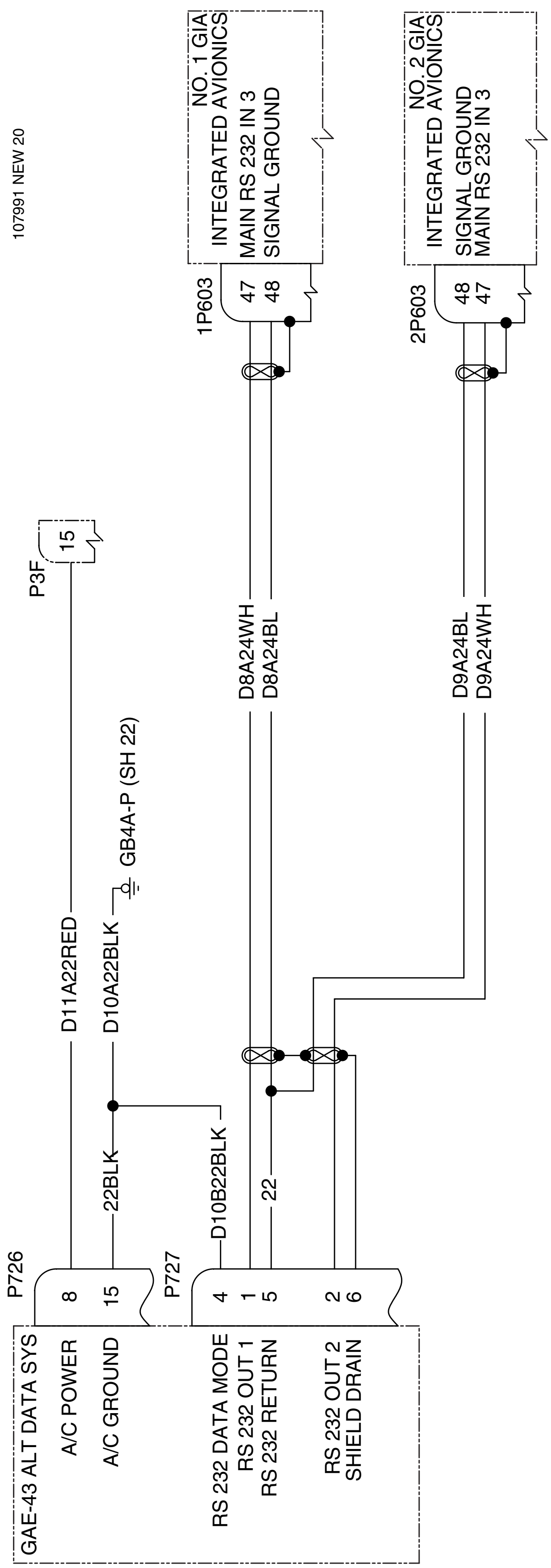
Cabin Pressurization
Figure 1 (Sheet 7 of 10)



Cabin Pressurization
Figure 1 (Sheet 8 of 10)

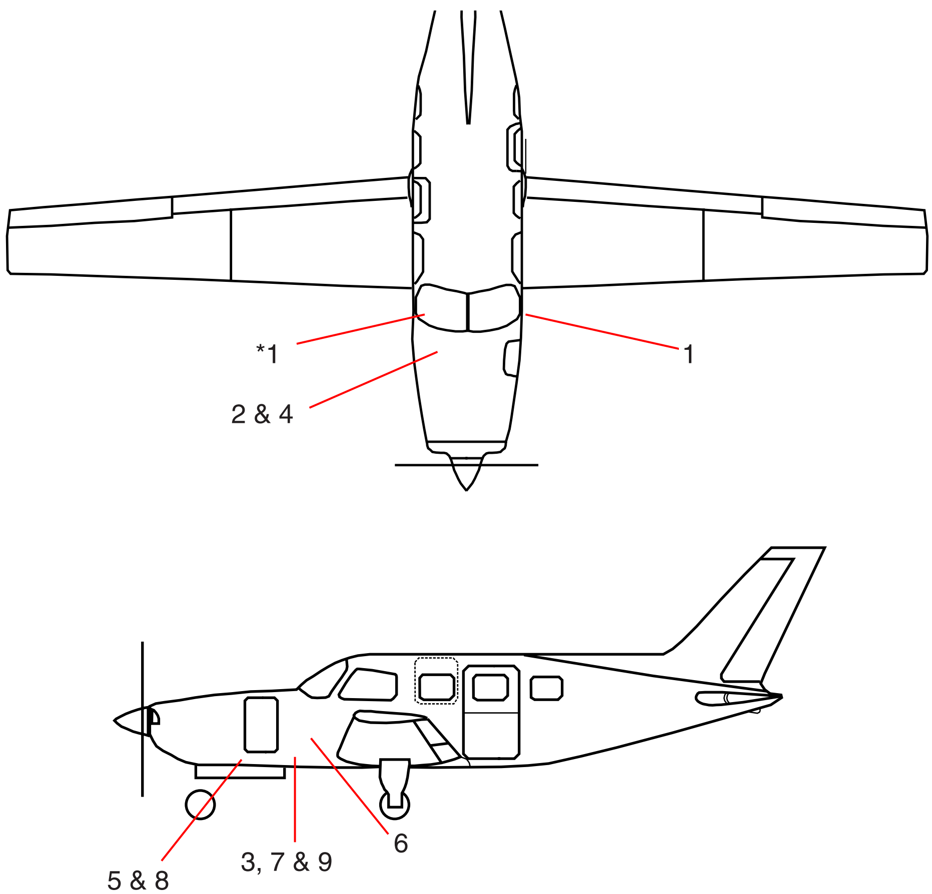


Cabin Pressurization
Figure 1 (Sheet 9 of 10)

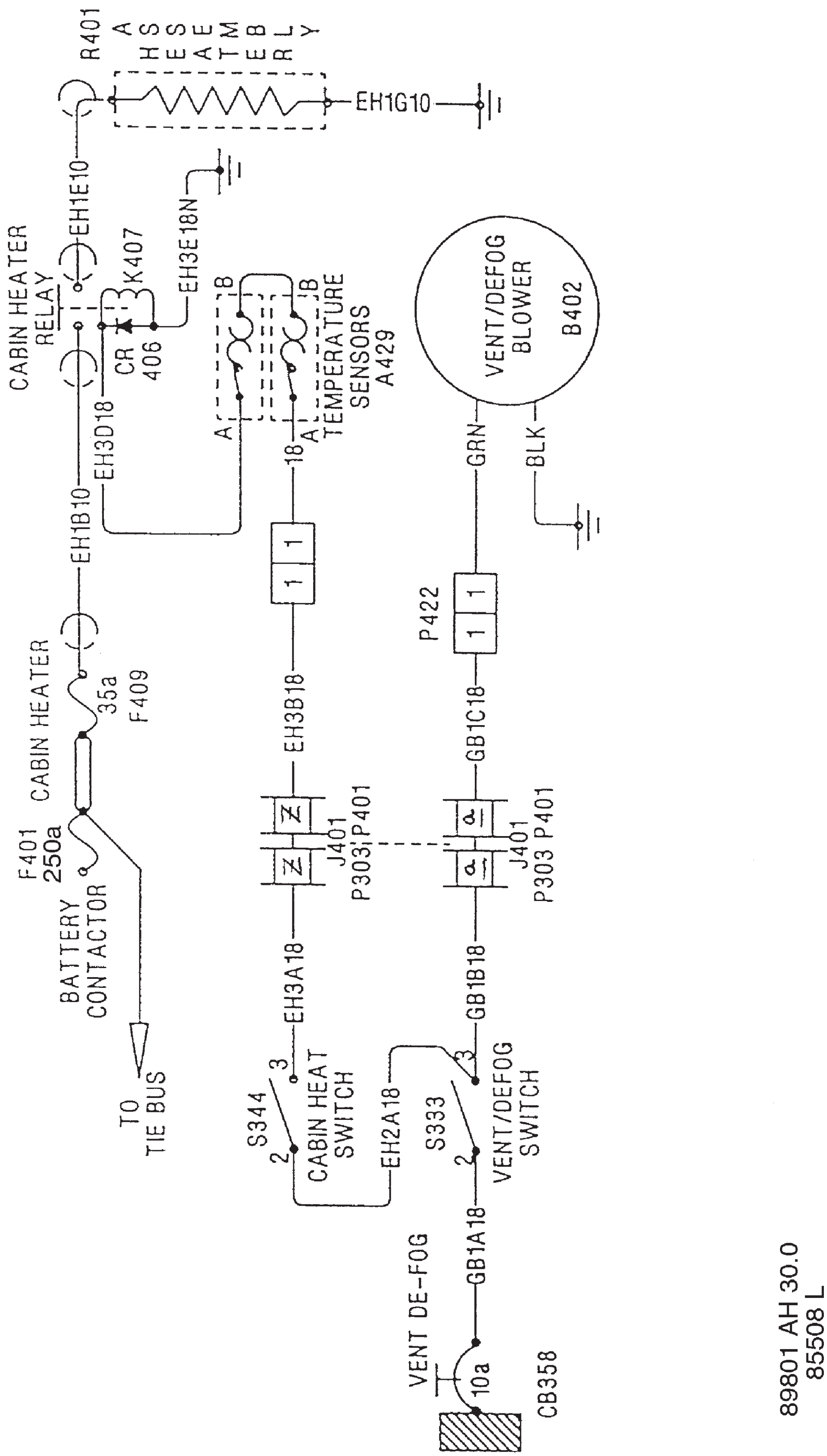


Cabin Pressurization
Figure 1 (Sheet 10 of 10)

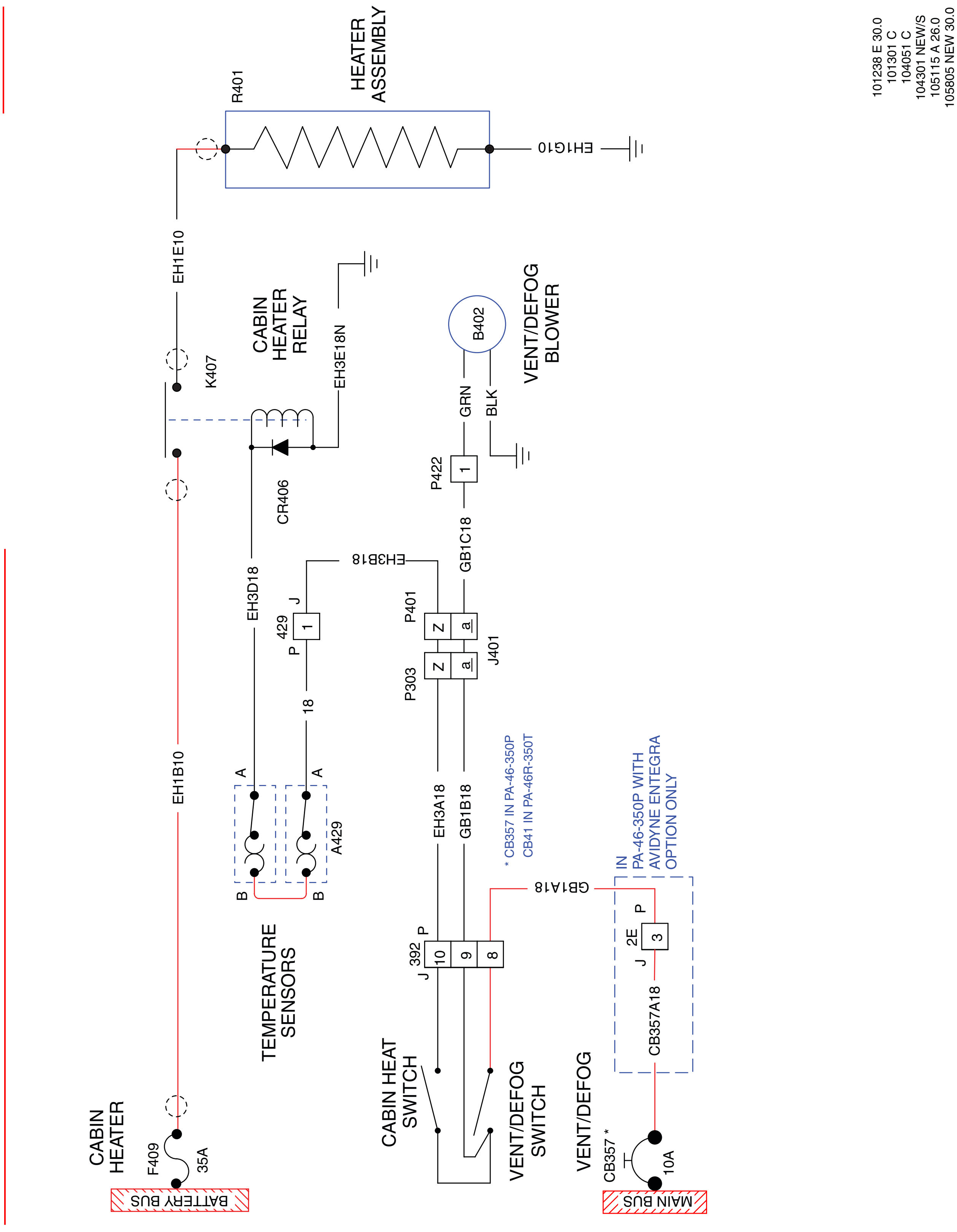
Item #	Designation	Description
1	CB358, CB357, CB41*	Circuit Breaker - Vent / Defog (10 Amp)
2	S333	Vent/Defog Switch
3	B402	Vent/Defog Blower
4	S344	Cabin Heat Switch
5	F401	100 Amp Current Limiter
6	K407	Cabin Heater Relay
7	A429	Temperature Sensors
8	F409	Current Limiter (35 Amp)
9	R401	Heater Assembly



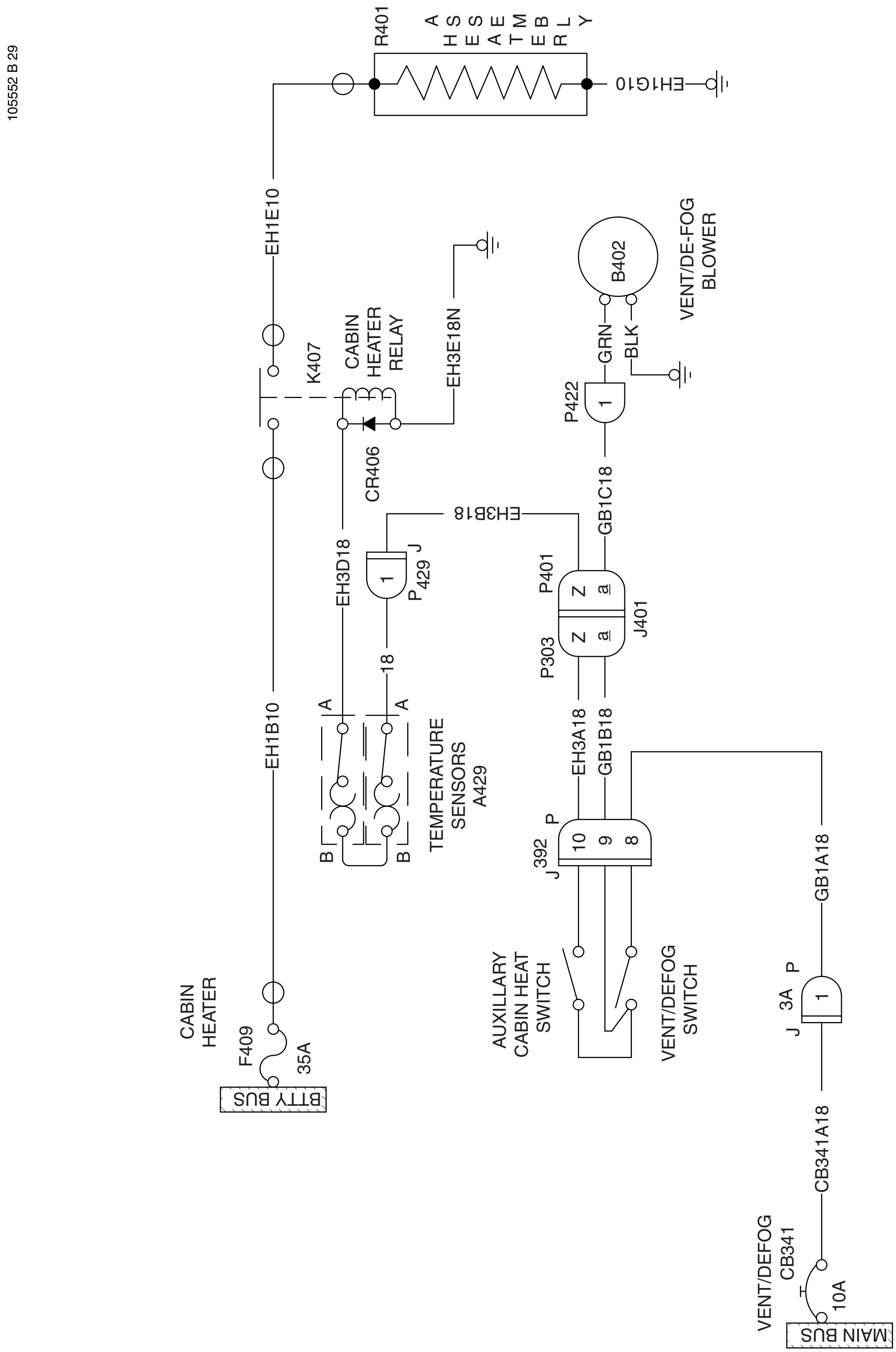
Vent Defogger and Cabin Heater
Figure 1 (Sheet 1 of 5)



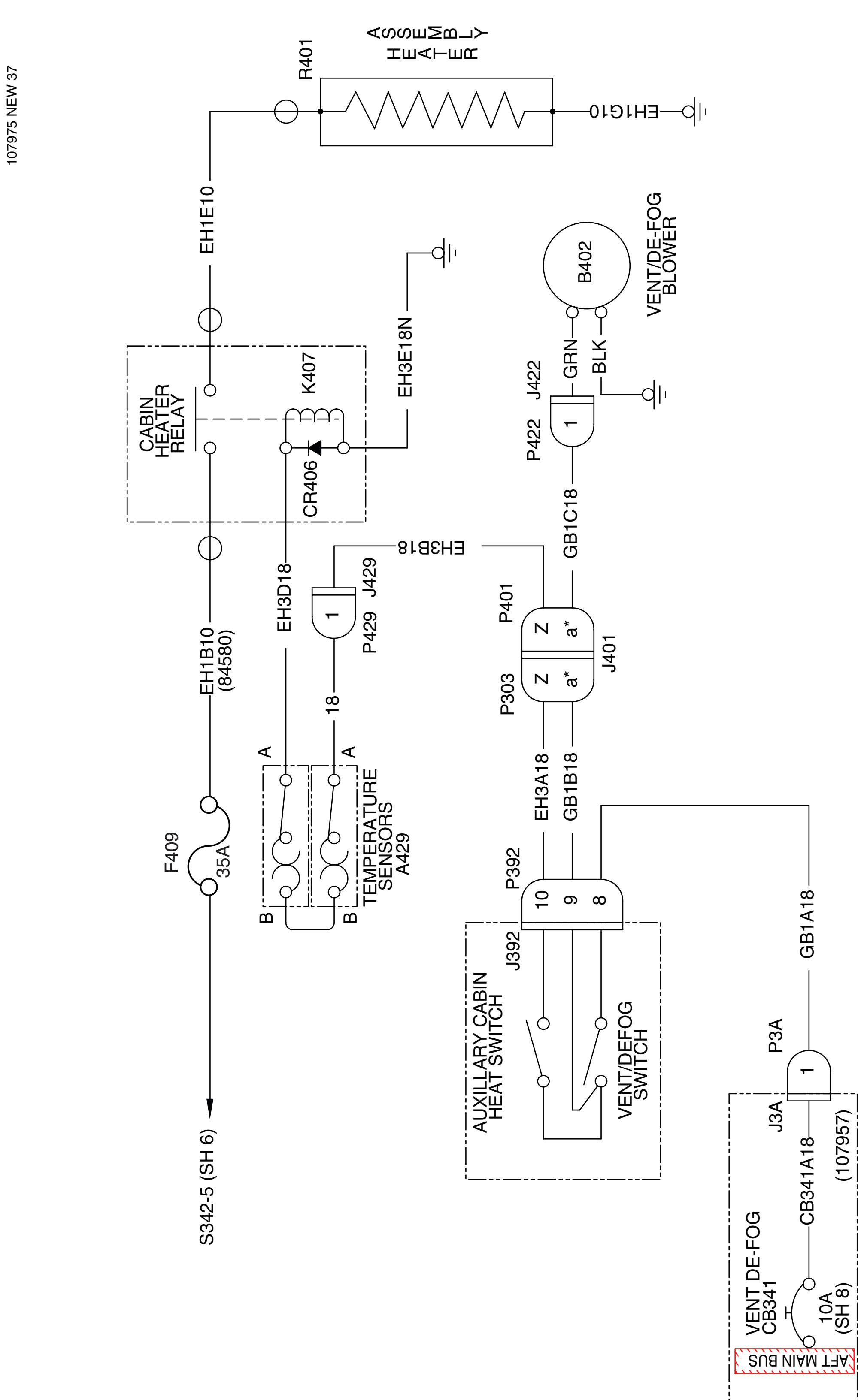
Vent Defogger and Cabin Heater
Figure 1 (Sheet 2 of 5)



Vent Defogger and Cabin Heater
Figure 1 (Sheet 3 of 5)



Vent Defogger and Cabin Heater
Figure 1 (Sheet 4 of 5)



Vent Defogger and Cabin Heater
Figure 1 (Sheet 5 of 5)

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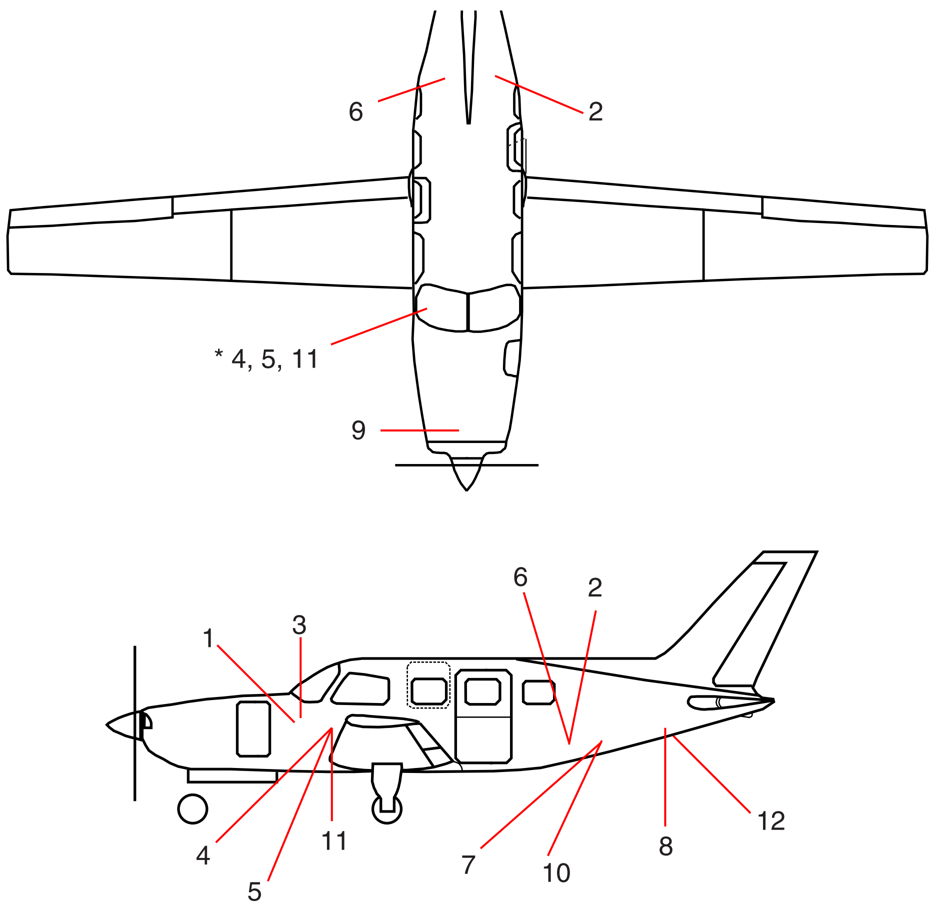
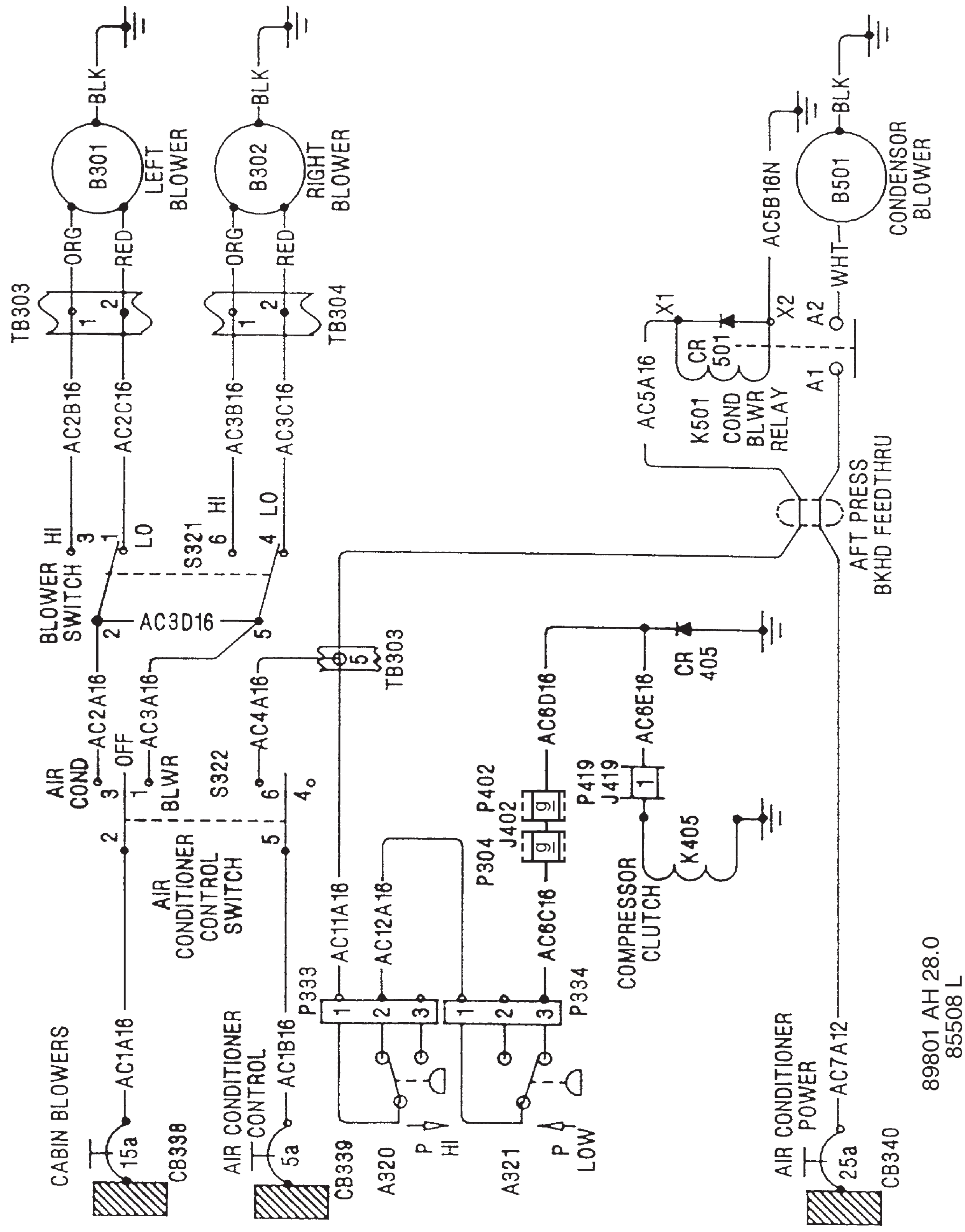
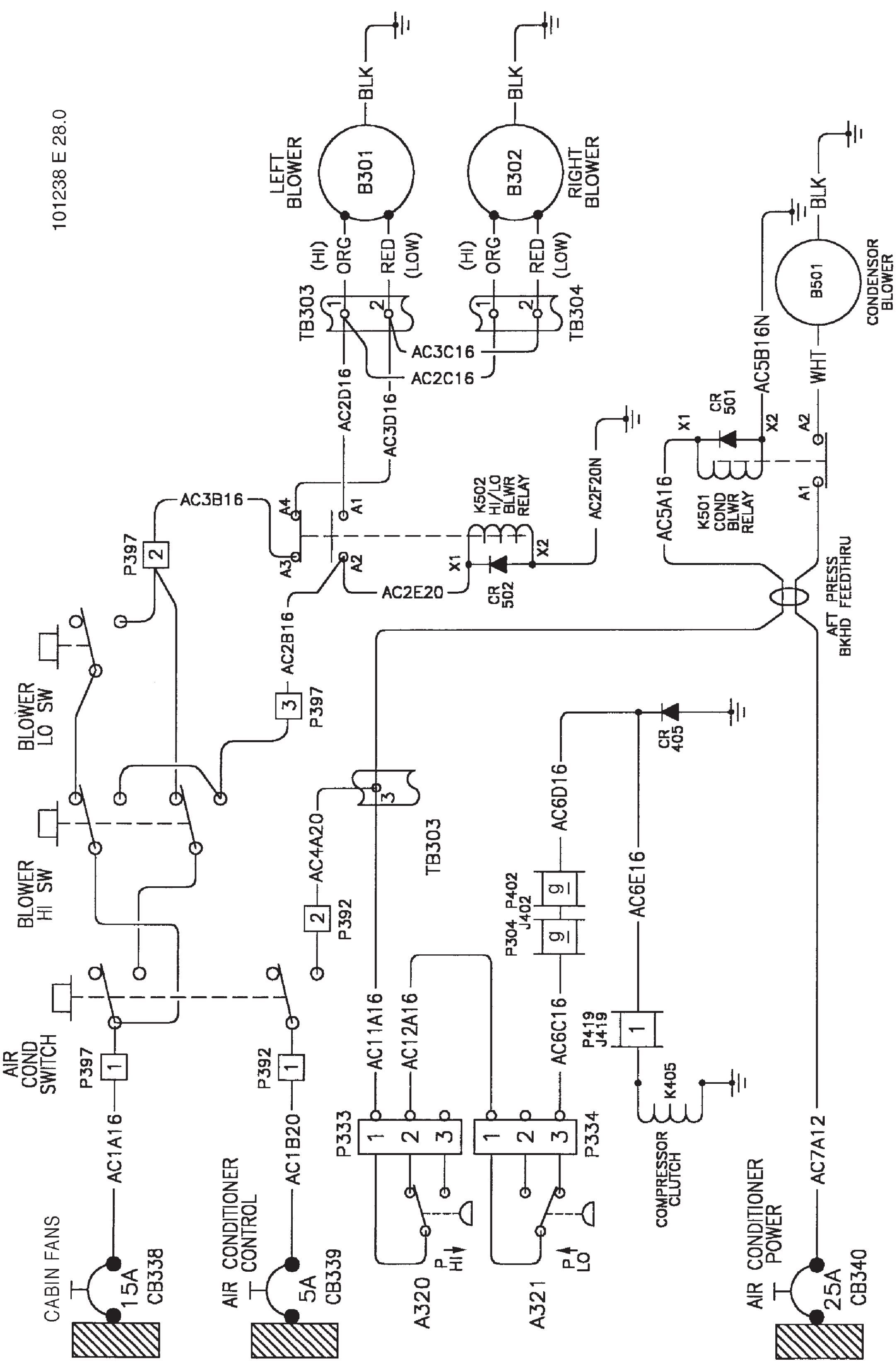
Item #	Designation	Description
1	S321	Recirculation Blower Switch
2	B301	Left Blower Fan Motor
3	S322	Air Conditioner Switch
4	CB338, CB57*	Circuit Breaker - Cabin Blowers/Fans CB (15 Amp)
5	CB339, CB58*	Circuit Breaker - Air Conditioner Control (5 Amp)
6	B302	Right Blower Fan Motor
7	A320	High Pressure Switch
8	K501	Condenser Blower Relay
9	K405	Compressor Clutch Assembly
10	A321	Low Pressure Switch
11	CB340, CB59*	Circuit Breaker - Air Conditioner Power (25 Amp)
12	B501	Condenser Blower Motor Assembly
13	CR405	Diode
14	CR501	Diode



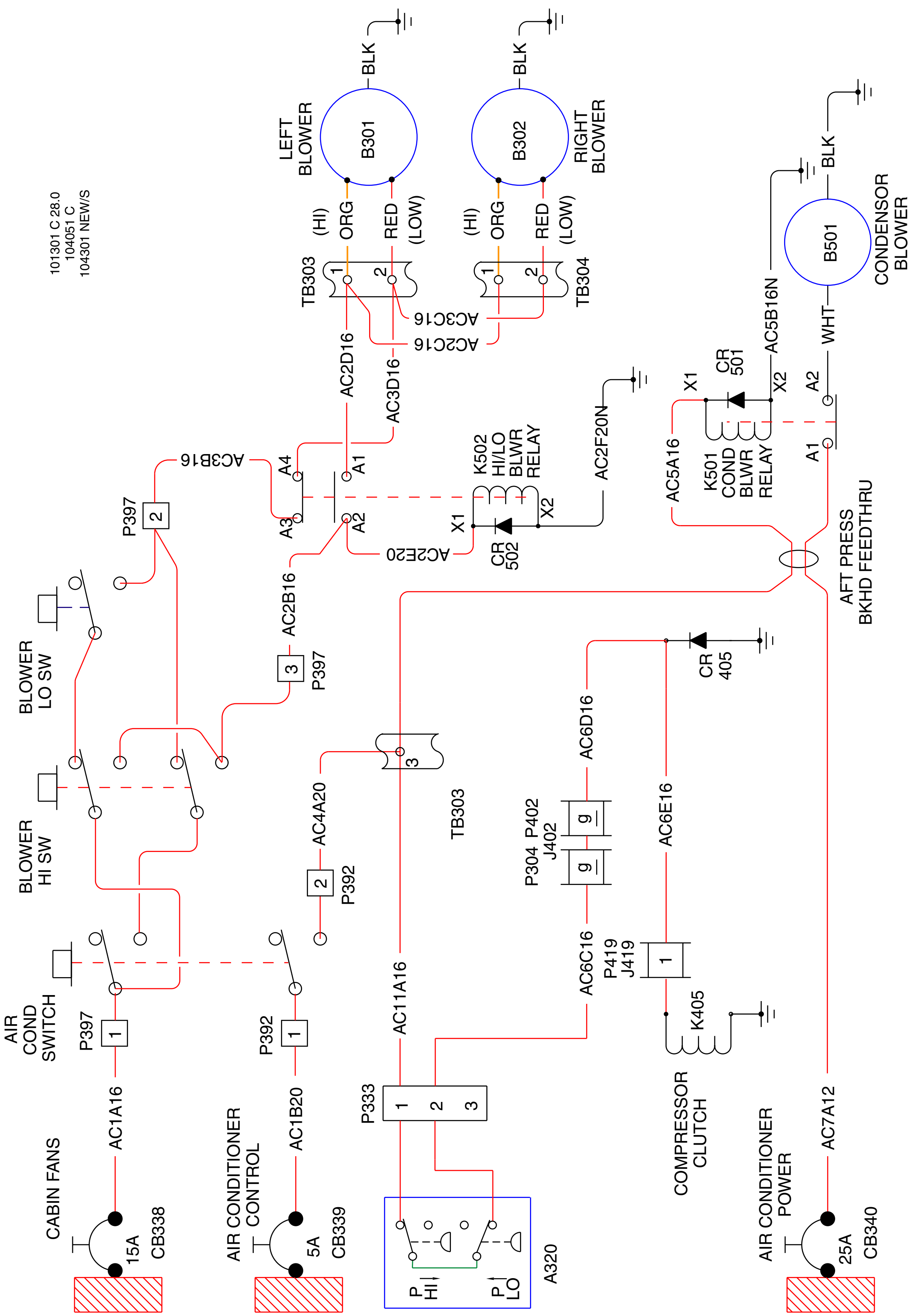
Air Conditioning and Blowers
Figure 1 (Sheet 1 of 7)



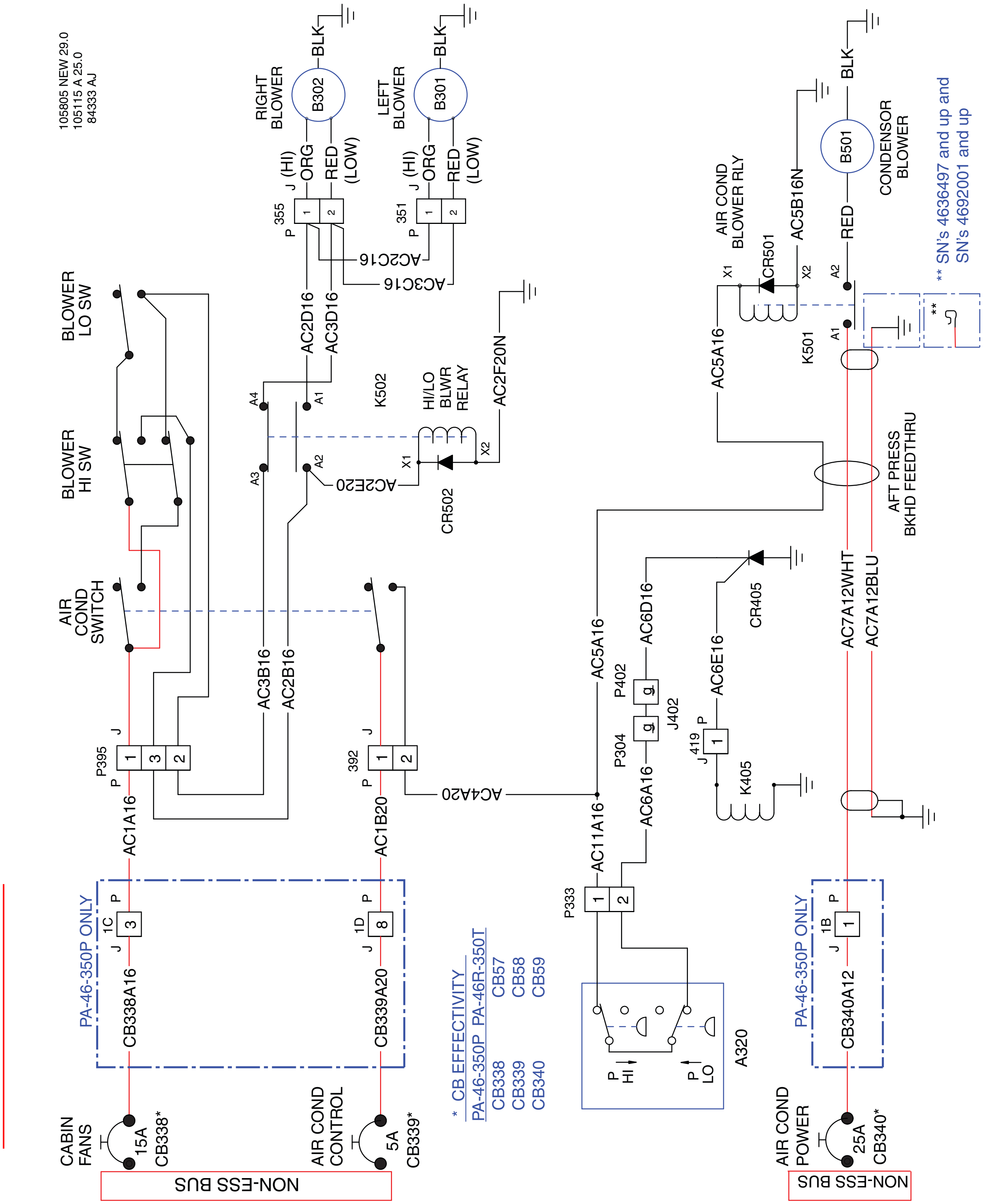
Air Conditioning and Blowers
Figure 1 (Sheet 2 of 7)



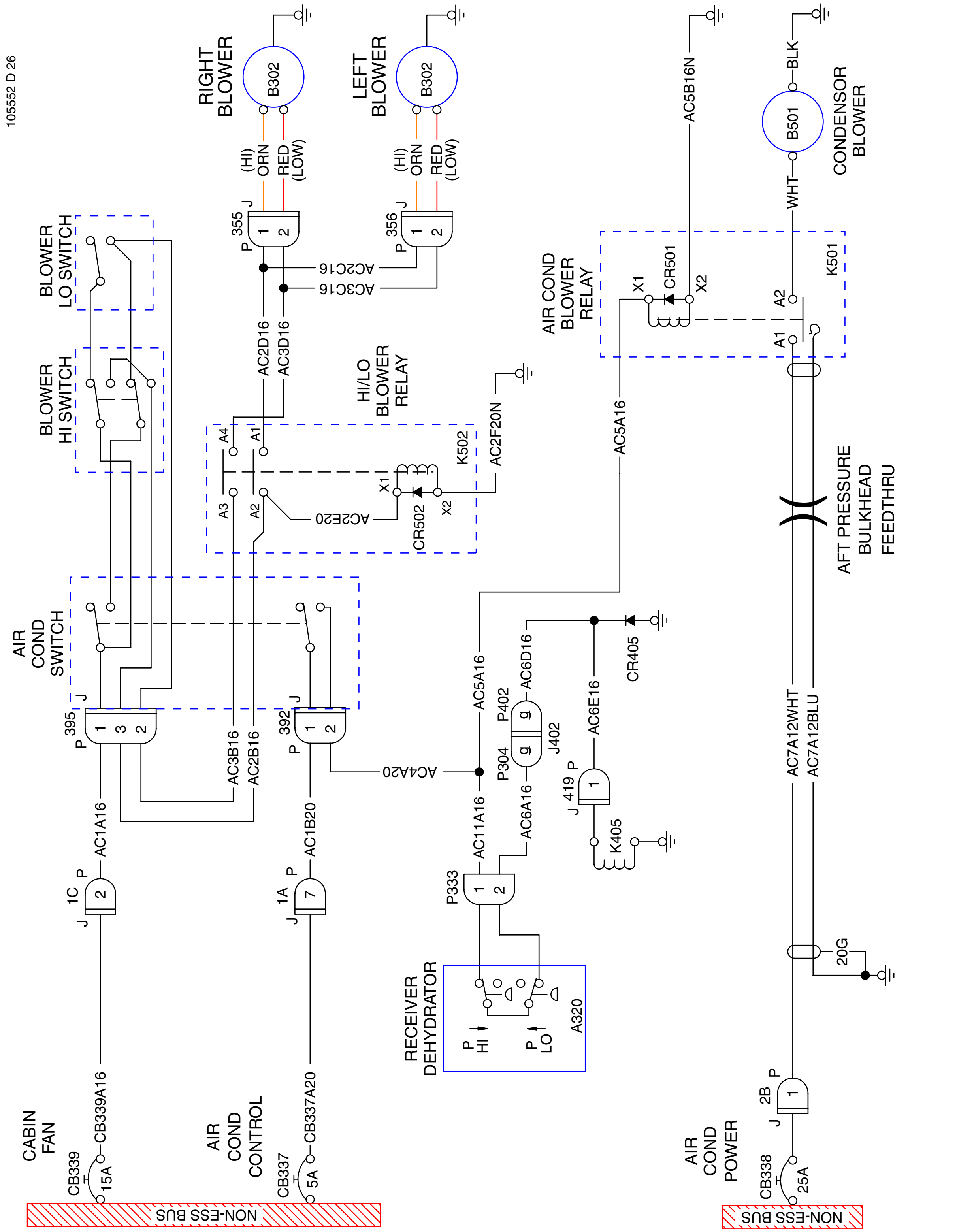
Air Conditioning and Blowers
Figure 1 (Sheet 3 of 7)



Air Conditioning and Blowers
Figure 1 (Sheet 4 of 7)

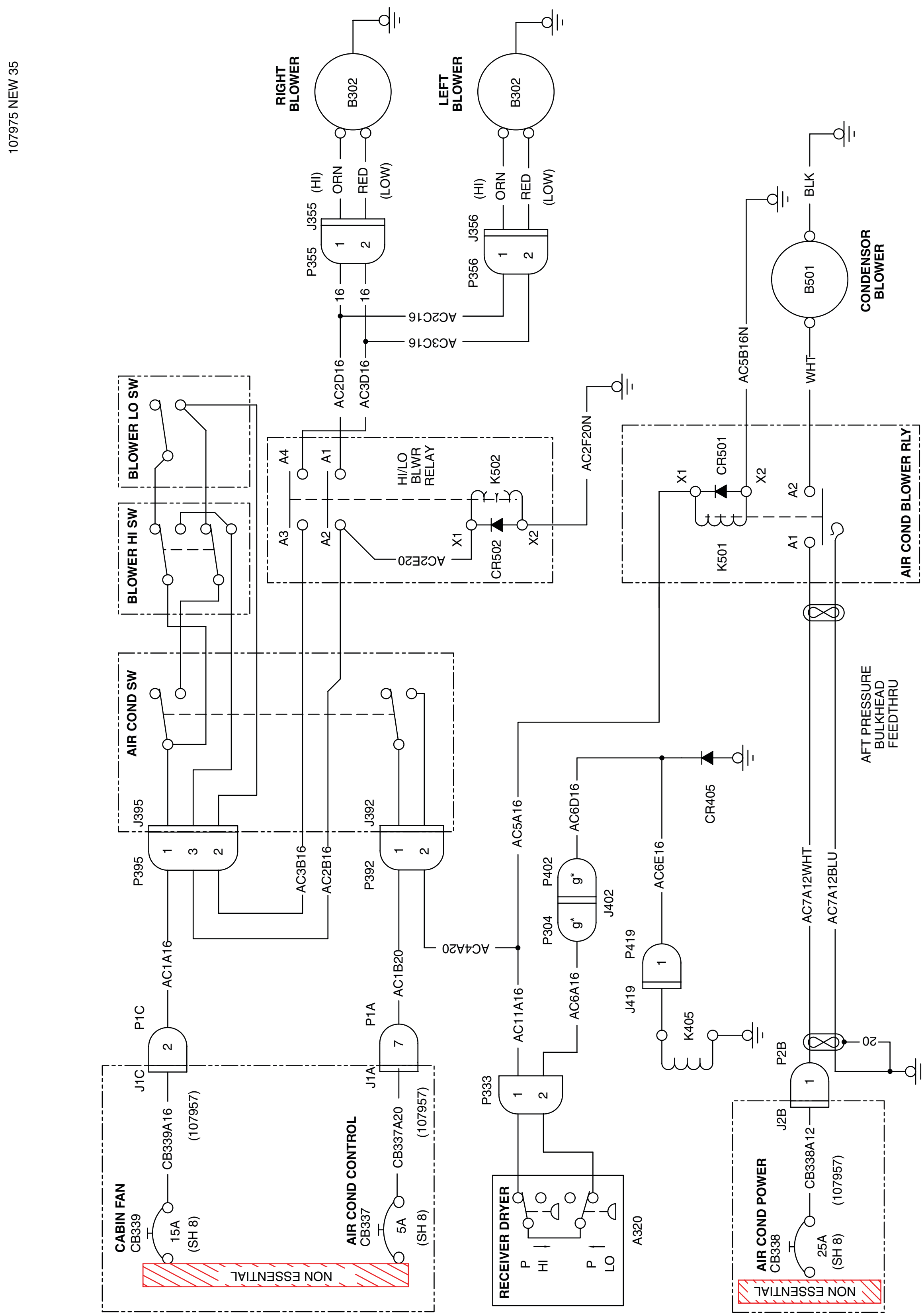


Air Conditioning and Blowers
Figure 1 (Sheet 5 of 7)



Air Conditioning and Blowers
Figure 1 (Sheet 6 of 7)

Effectivity
4636460, 4636463–4636651, less 4636481 and 4636633
4692134 and up, less 4692141, 4692149, 4692153



Air Conditioning and Blowers
Figure 1 (Sheet 7 of 7)

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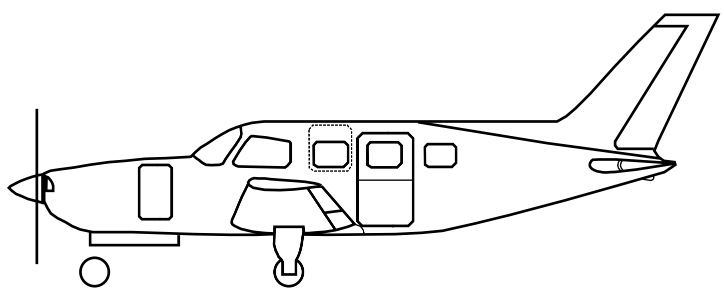
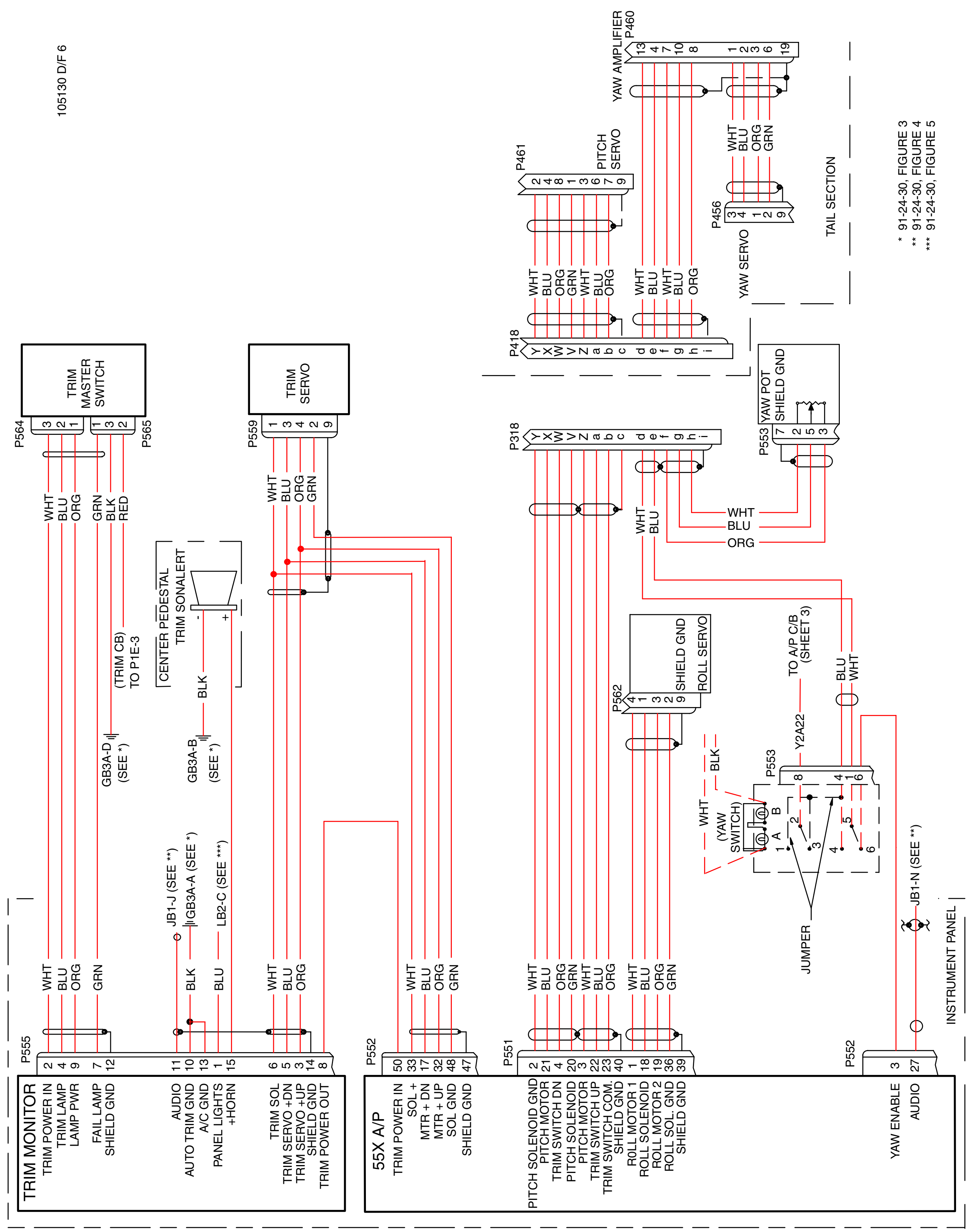
Item #	Designation	Description

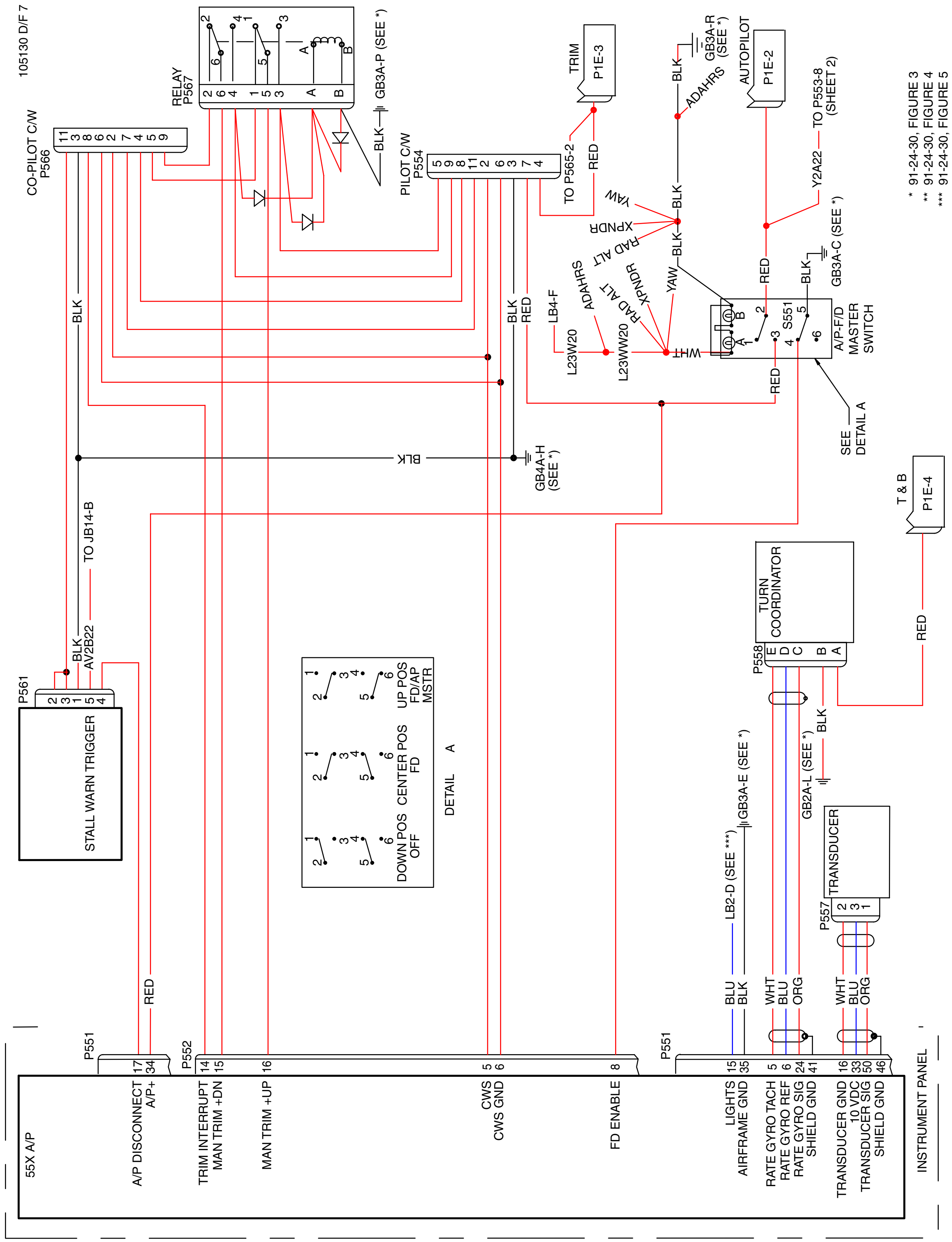
Information Pending - See Parts Catalog



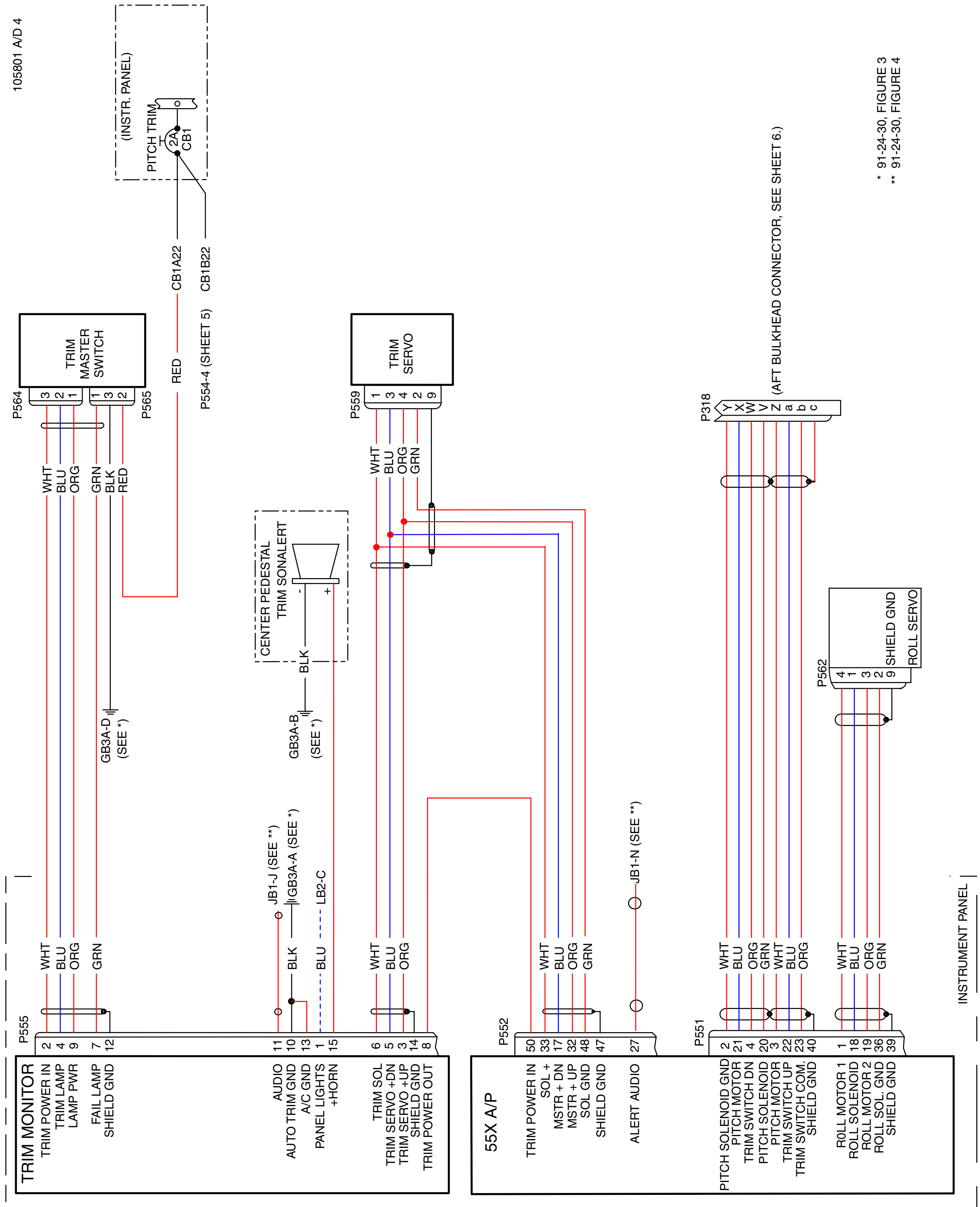
55X Autopilot
Figure 1 (Sheet 1 of 6)



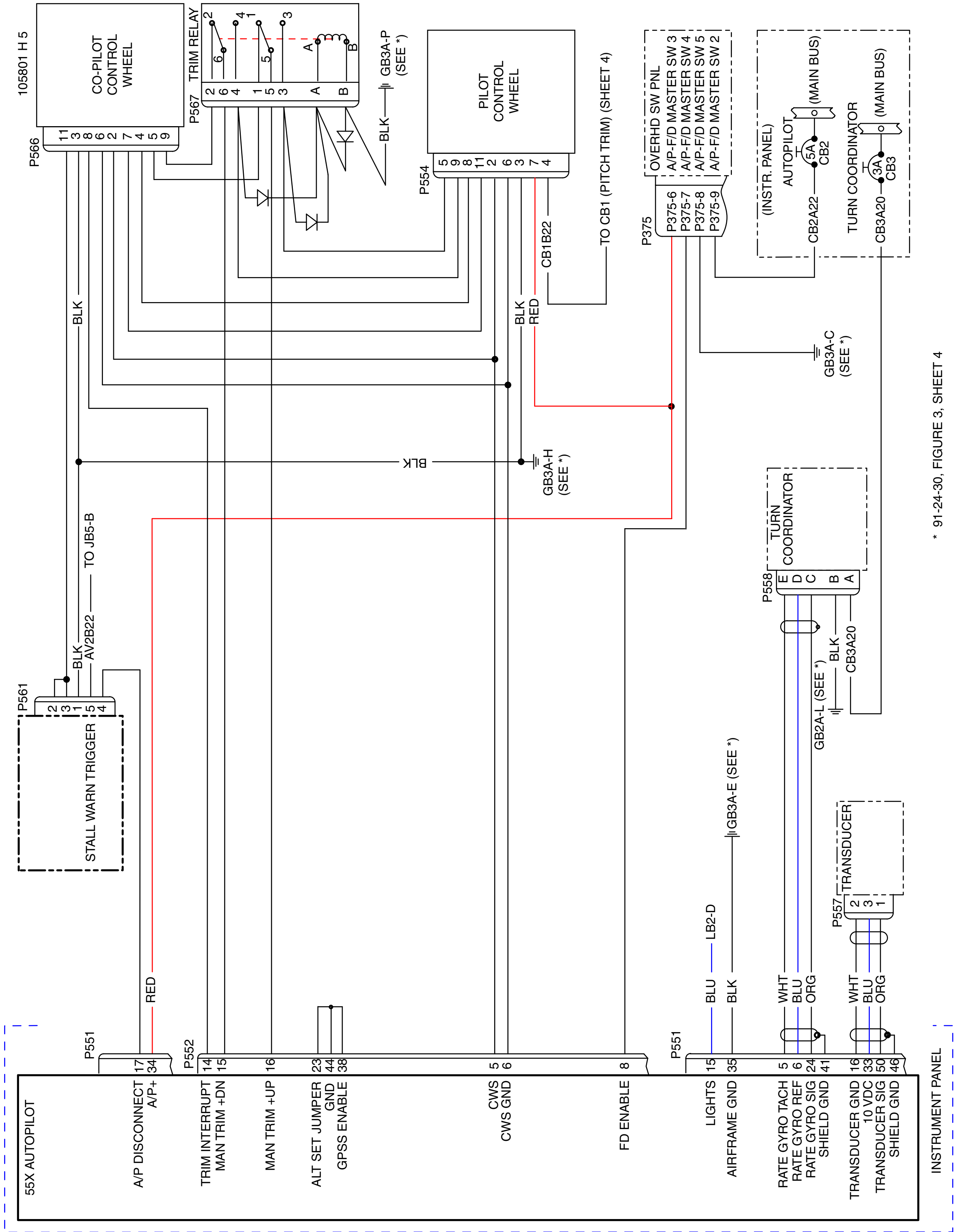
55X Autopilot
Figure 1 (Sheet 2 of 6)



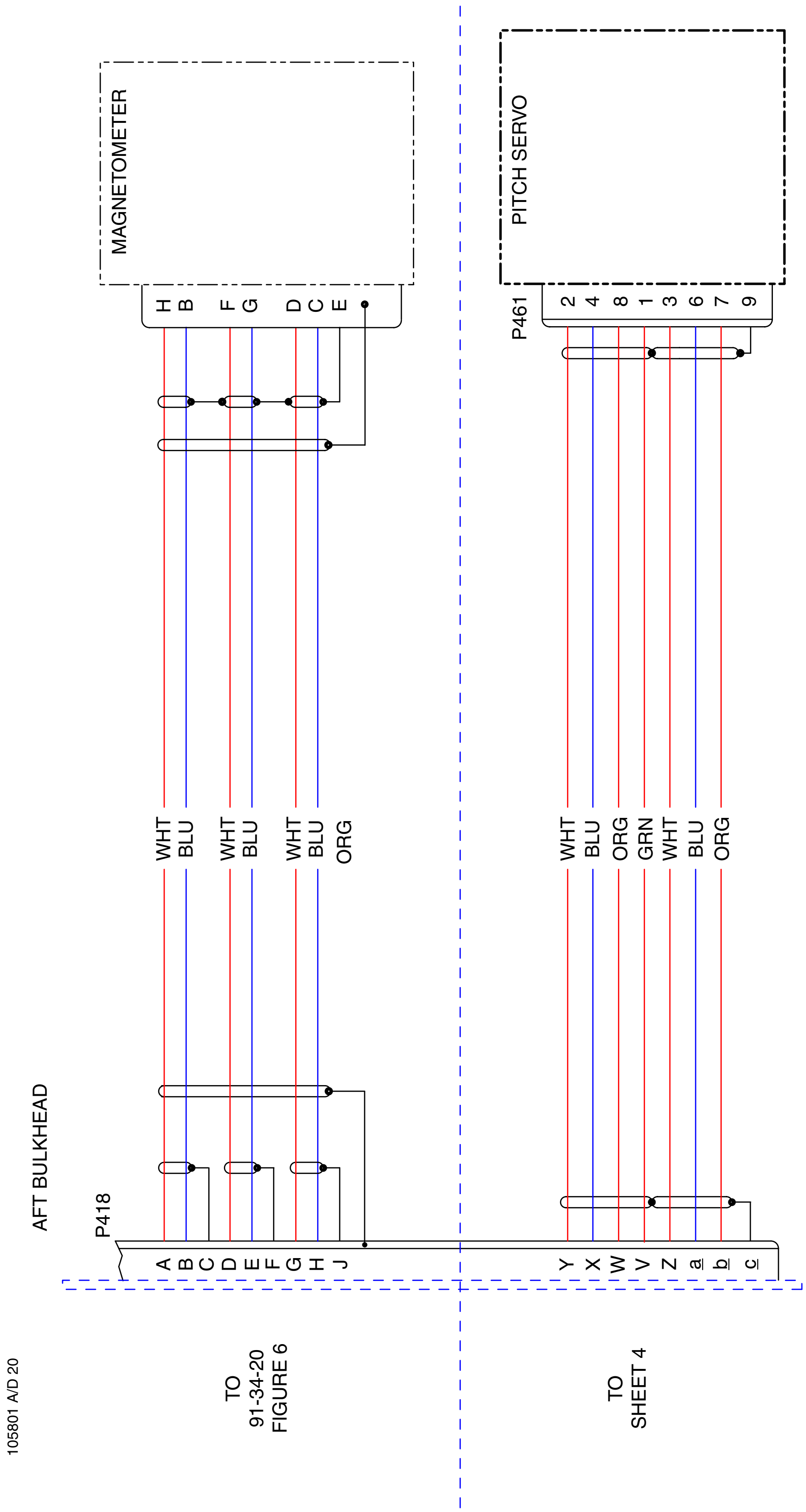
55X Autopilot
Figure 1 (Sheet 3 of 6)



55X Autopilot
Figure 1 (Sheet 4 of 6)



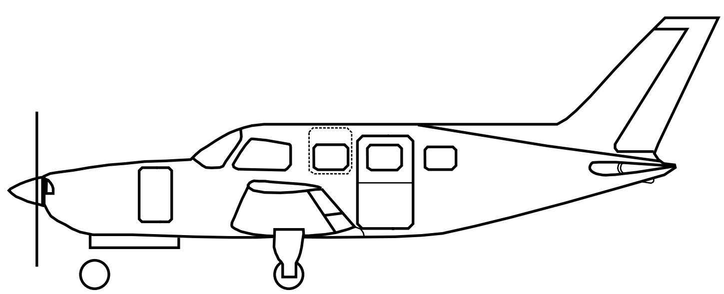
55X Autopilot
Figure 1 (Sheet 5 of 6)



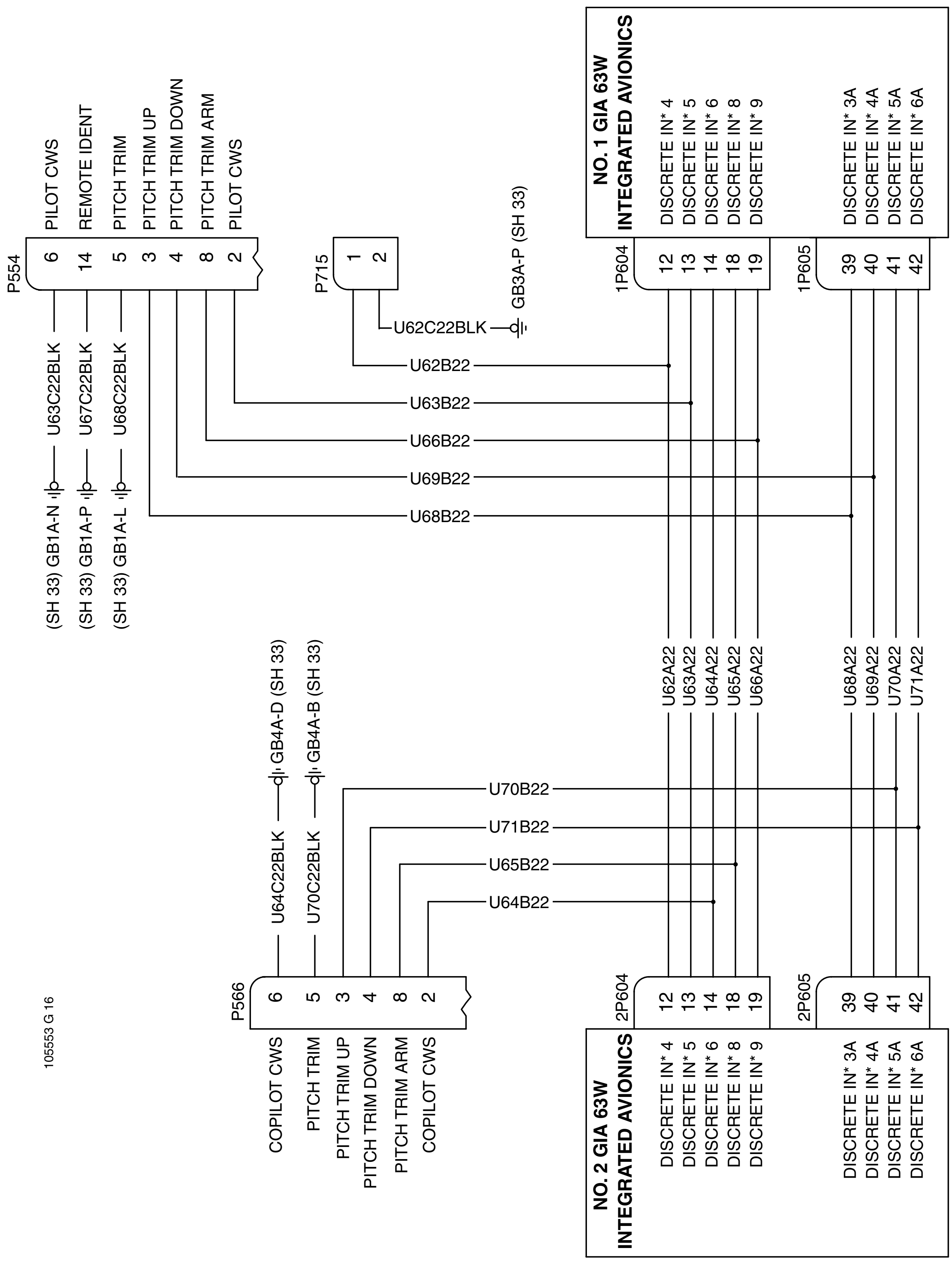
55X Autopilot
Figure 1 (Sheet 6 of 6)

Item #	Designation	Description

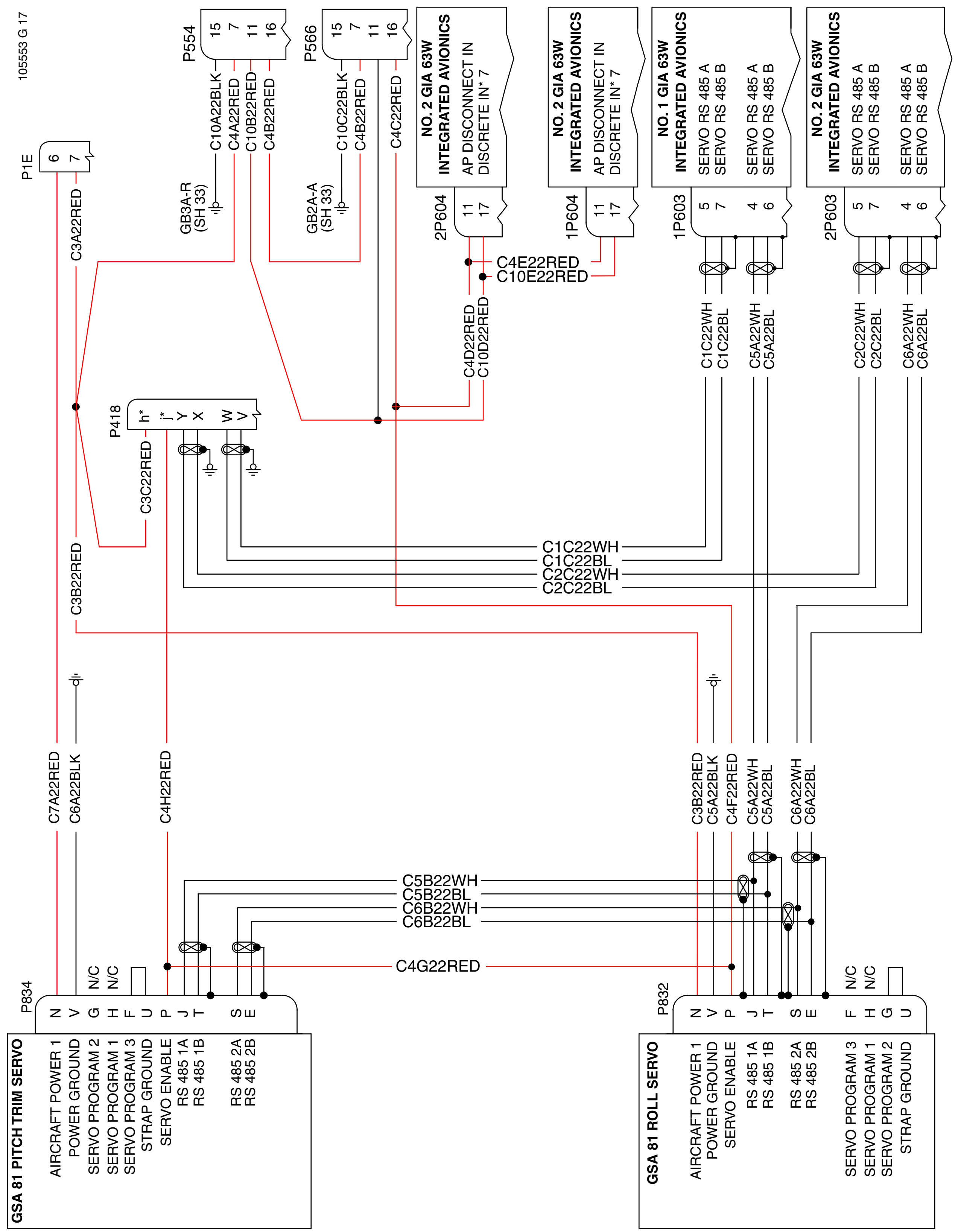
Information Pending - See Parts Catalog



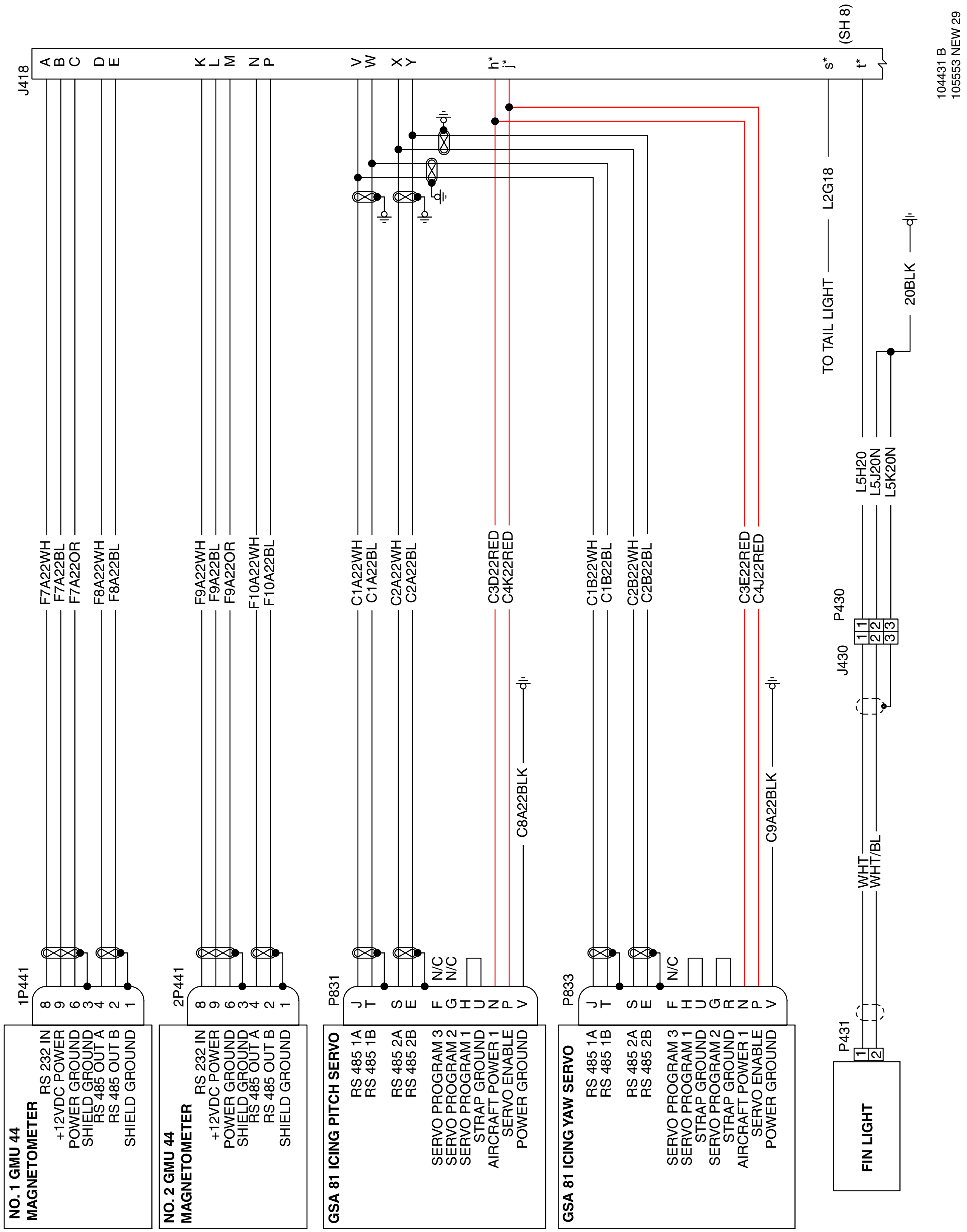
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 1 of 10)



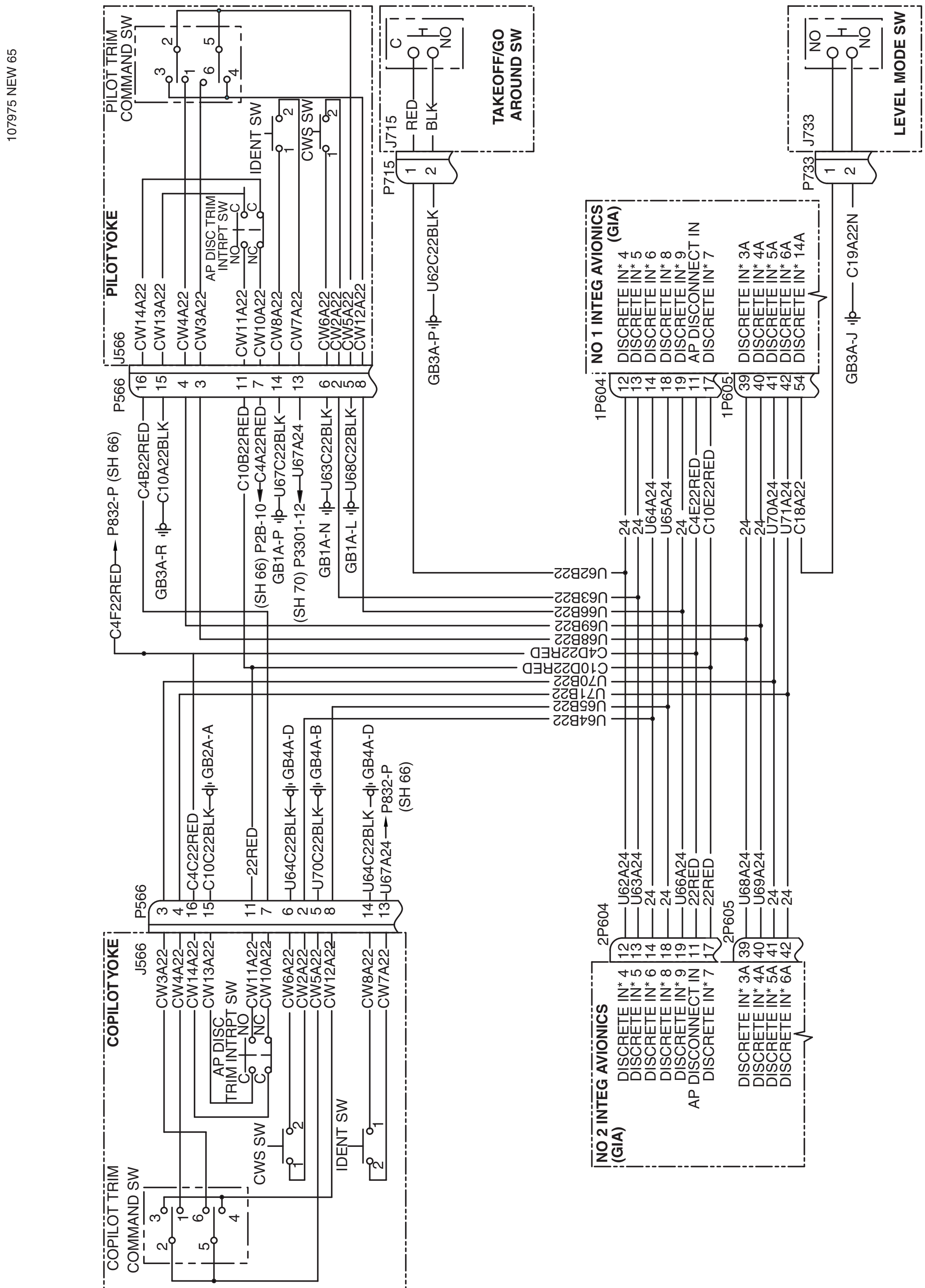
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 2 of 10)



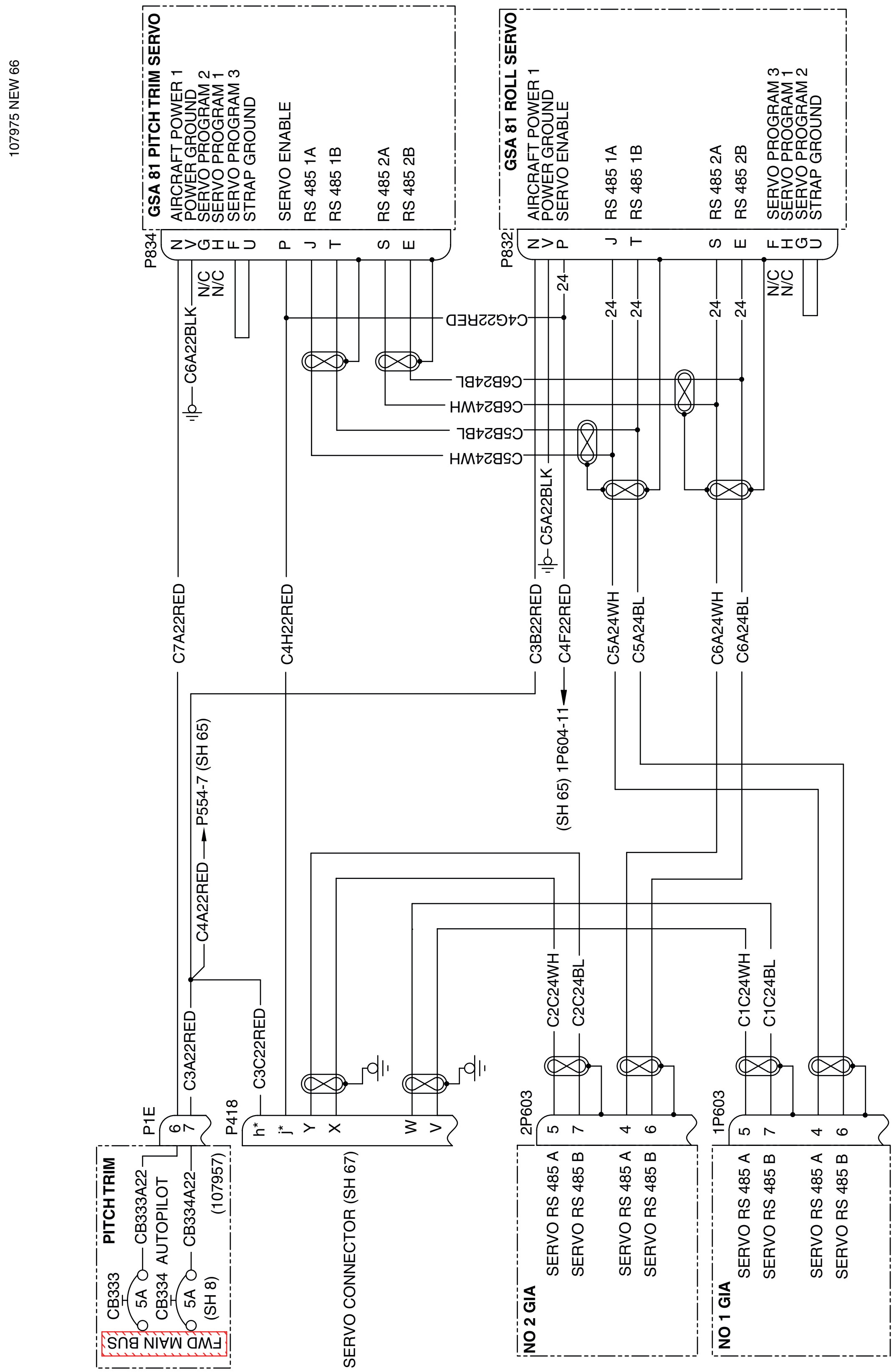
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 3 of 10)



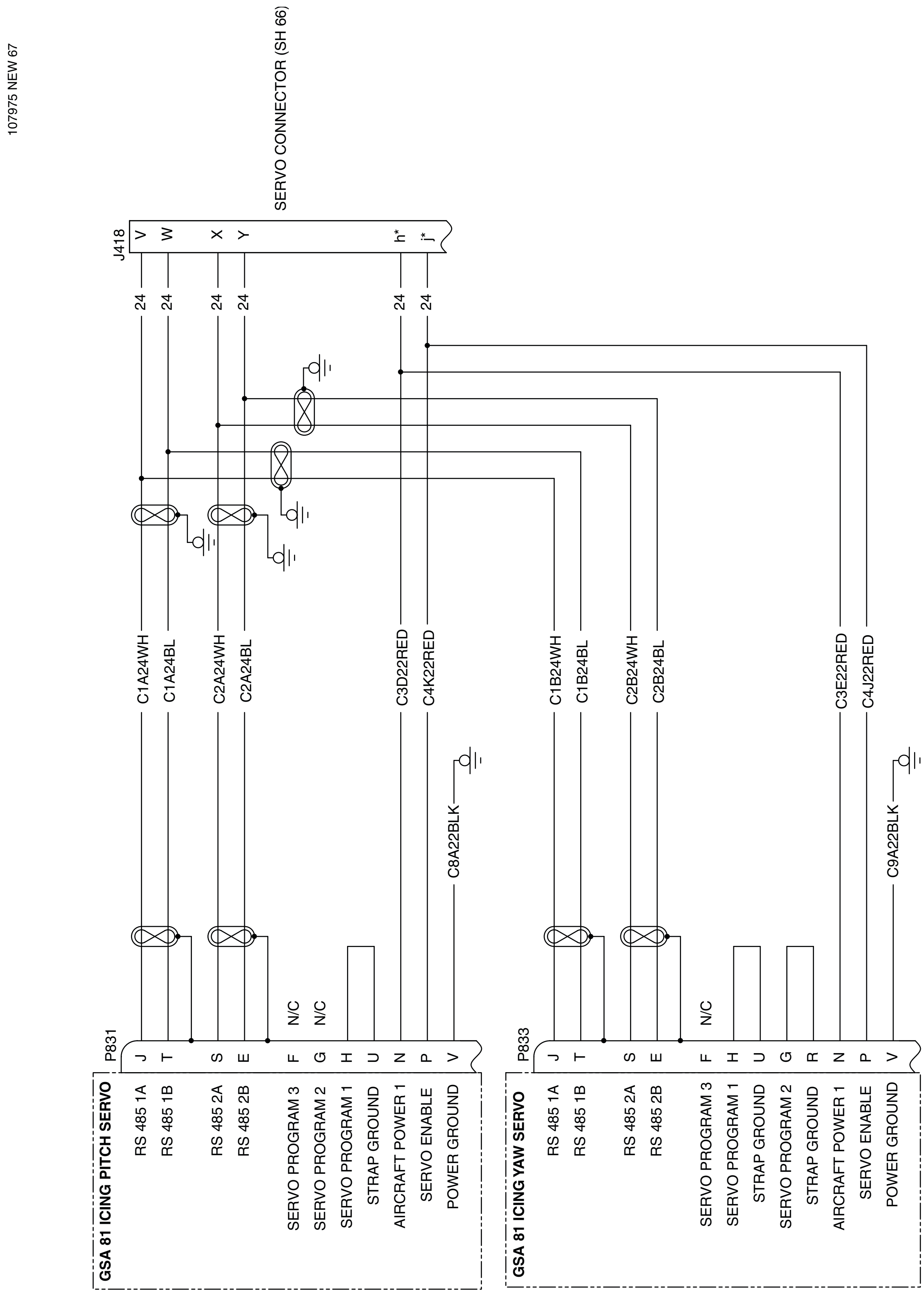
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 4 of 10)



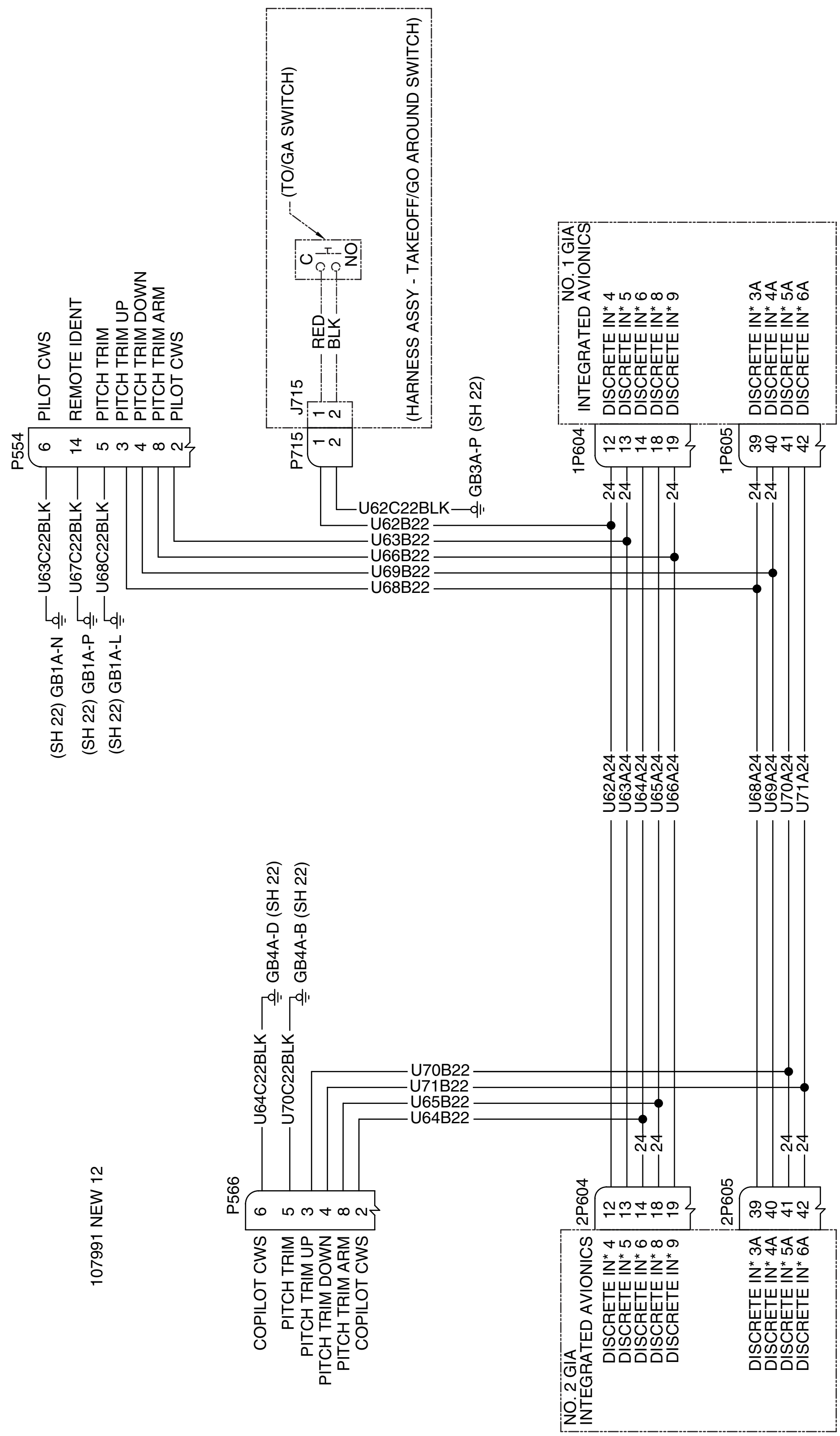
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 5 of 10)



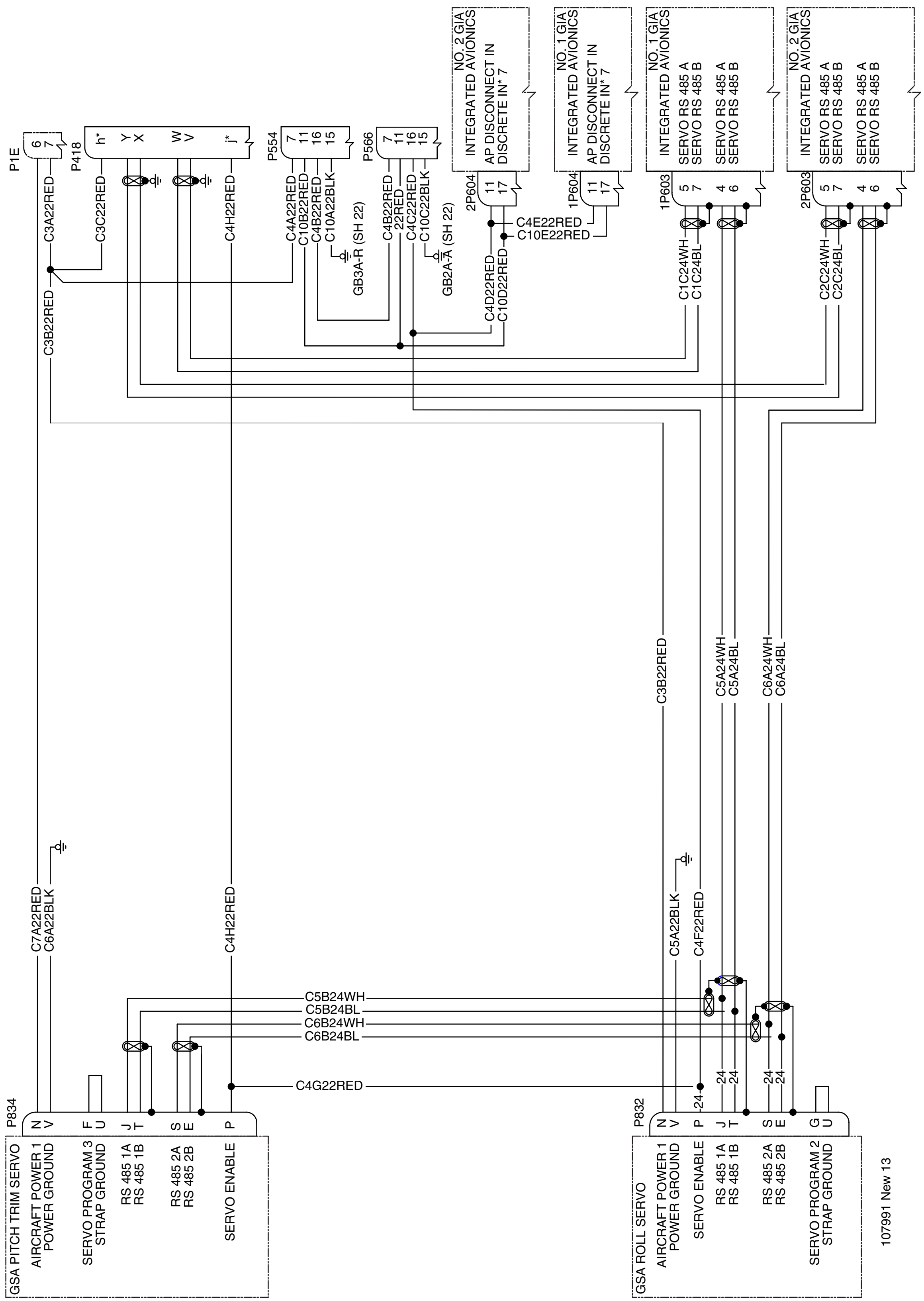
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 6 of 10)



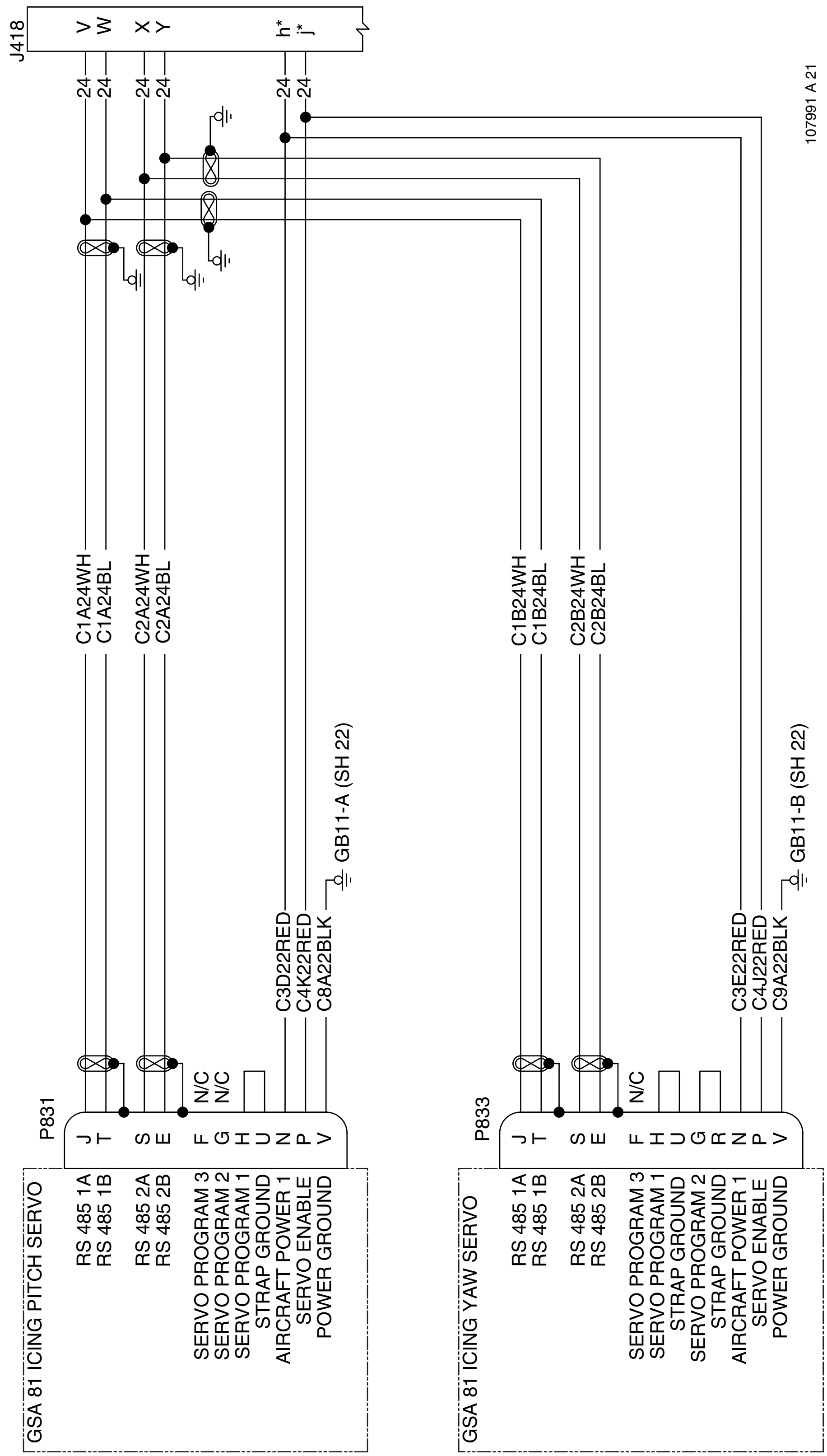
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 7 of 10)



GFC 700 Automatic Flight Control System
Figure 2 (Sheet 8 of 10)



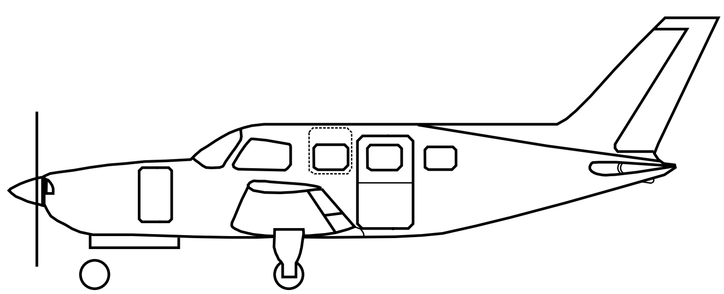
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 9 of 10)



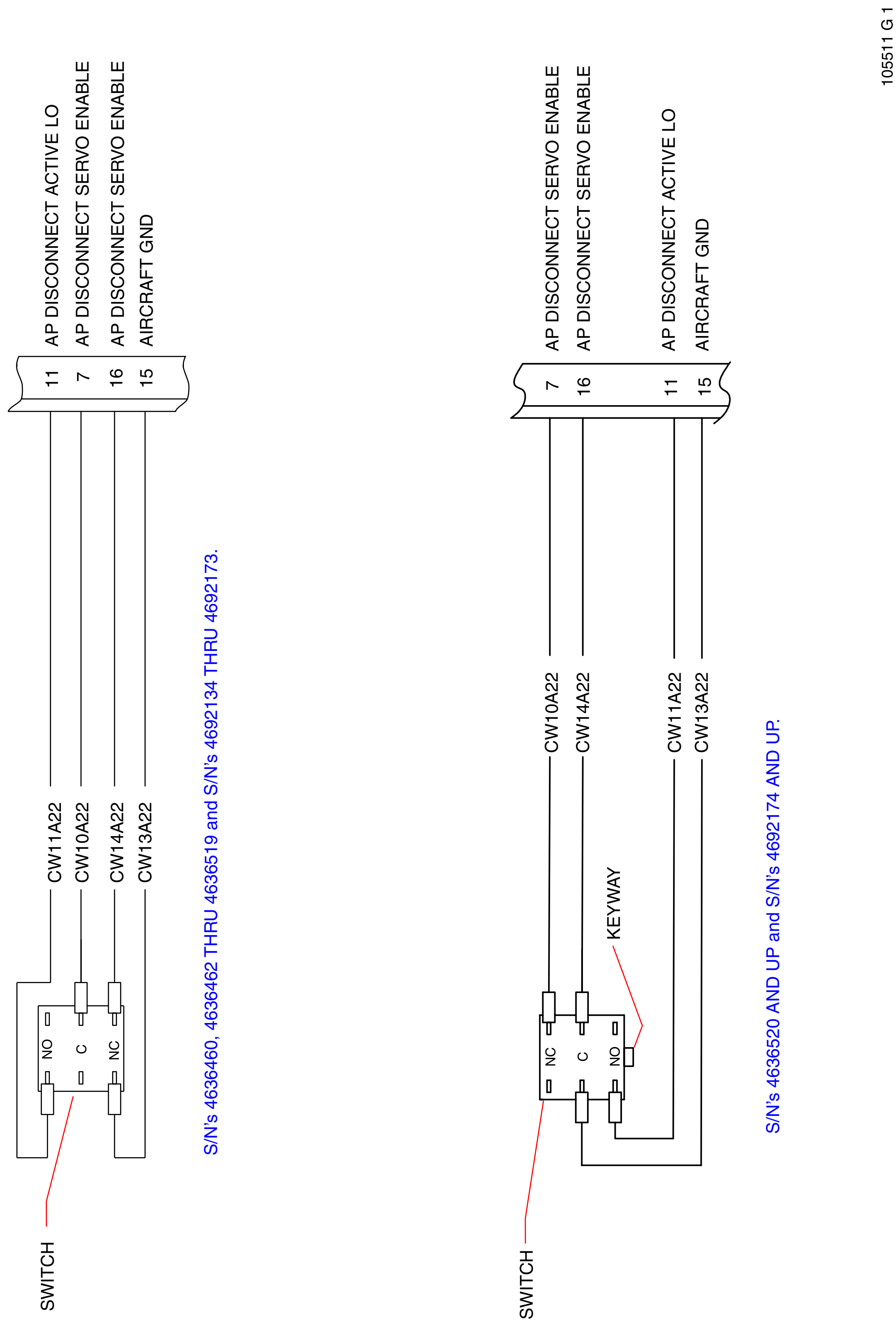
GFC 700 Automatic Flight Control System
Figure 2 (Sheet 10 of 10)

Item #	Designation	Description

Information Pending - See Parts Catalog



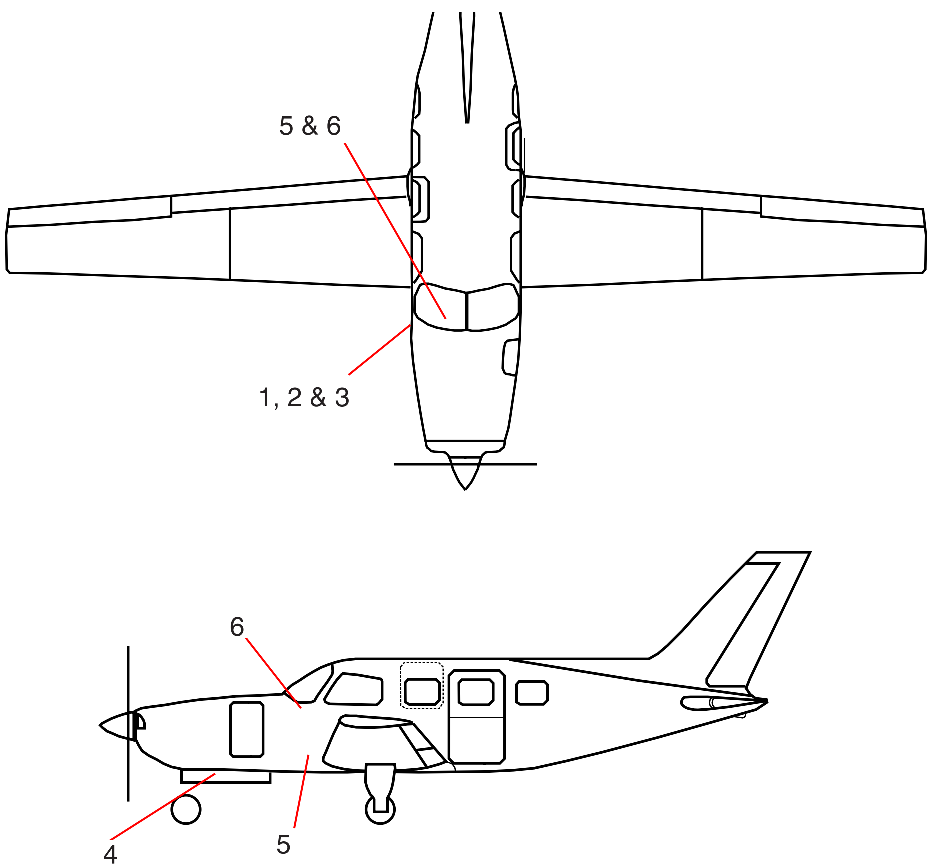
Control Wheel Switches
Figure 3 (Sheet 1 of 2)



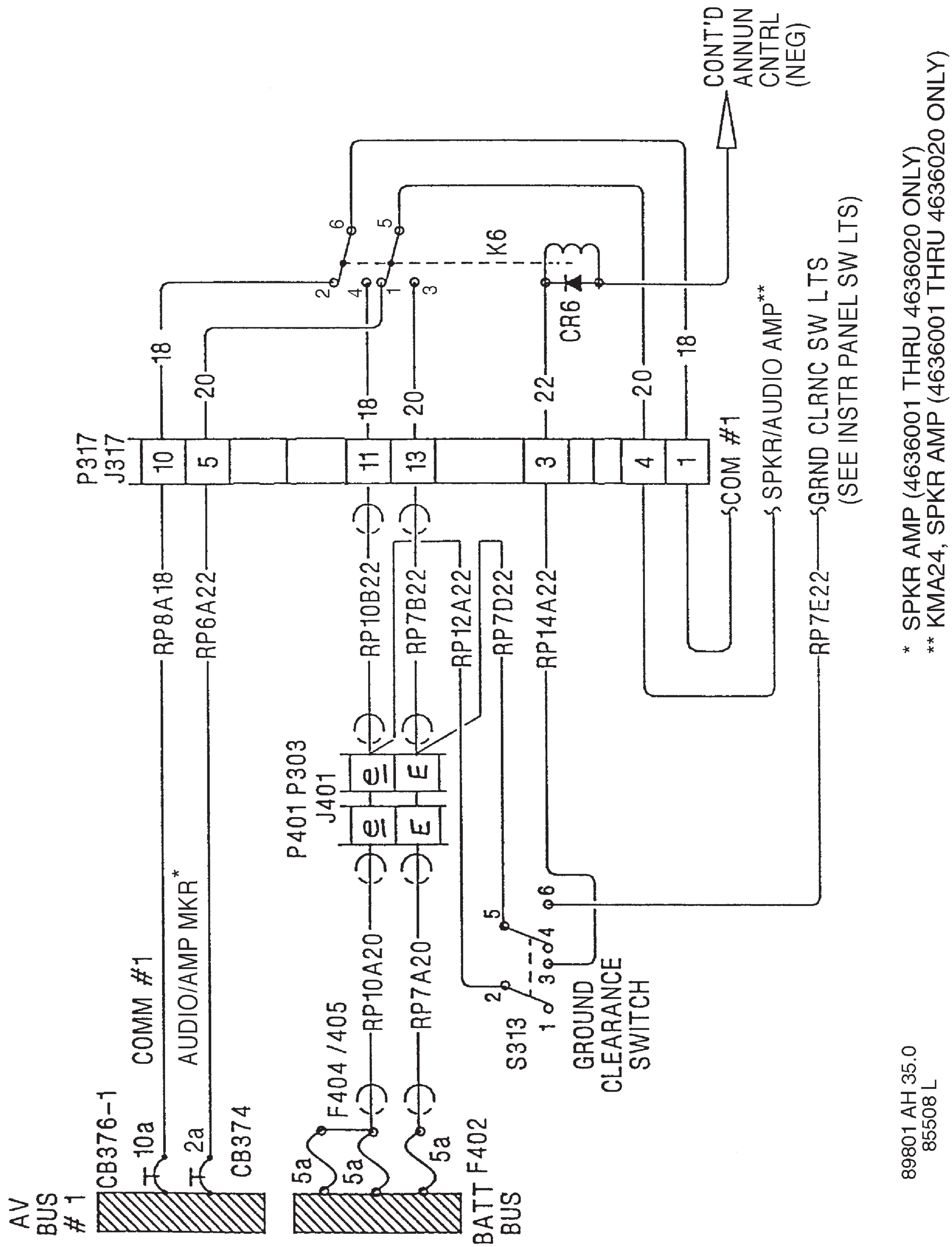
Control Wheel Switches
Figure 3 (Sheet 2 of 2)

Item #	Designation	Description
1	CB376-1, CB376, CB35	Circuit Breaker - Comm #1 (10 Amp)
2	CB374, CB373, CB32	Circuit Breaker - Audio/Amp Mkr (2 Amp) *
3	CB375-1, CB375, CB34	Circuit Breaker - Nav/Gps 1 (5 Amp) **
4	F402	Fuse (5 Amp)
	F404	Fuse (5 Amp)
	F405	Fuse (5 Amp)
5	K6	Ground Clearance Relay
6	S313	Ground Clearance Switch
		Ground Clearance Switch Cap
	CR6	Diode

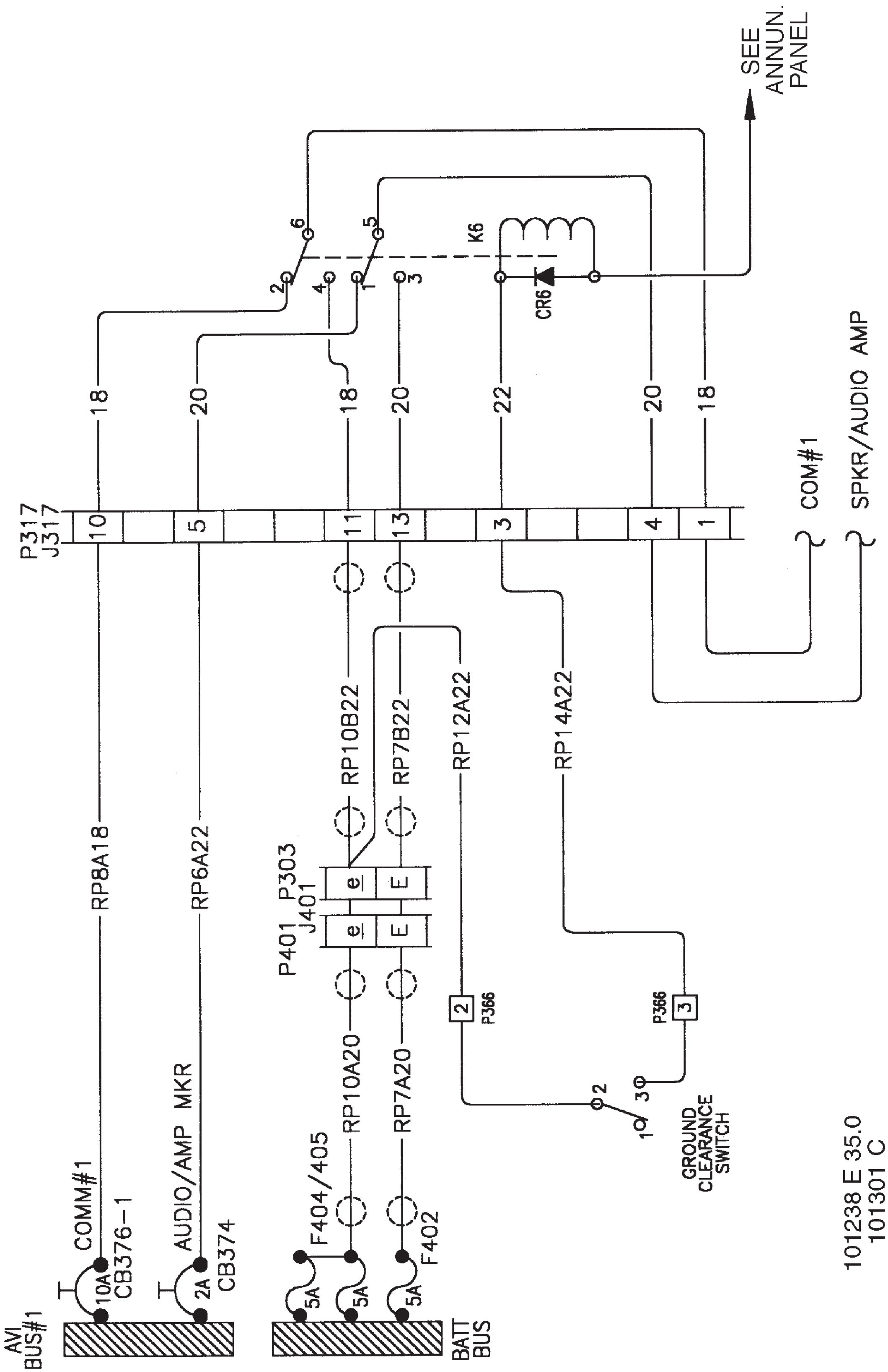
* 3 Amp Spkr Amplifier Circuit Breaker (4636001 thru 4636020 only)
2 Amp Circuit Breaker (4636021 thru 4636313 only)
5 Amp Circuit Breaker (4636314 and up, 4692001 and up)
** 5 Amp Circuit Breaker (4636248 and up, 4692001 and up)



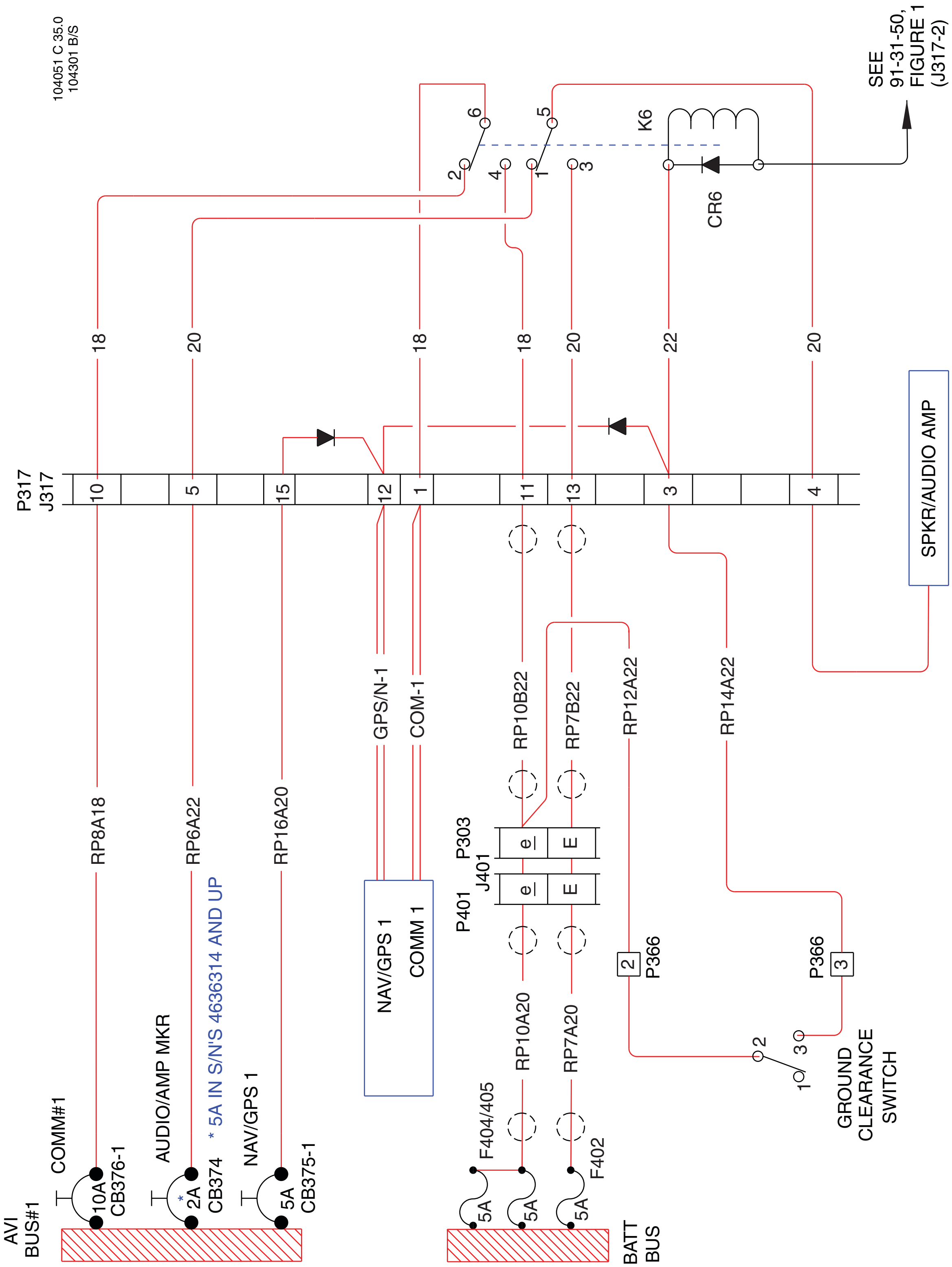
Ground Clearance
Figure 1 (Sheet 1 of 6)



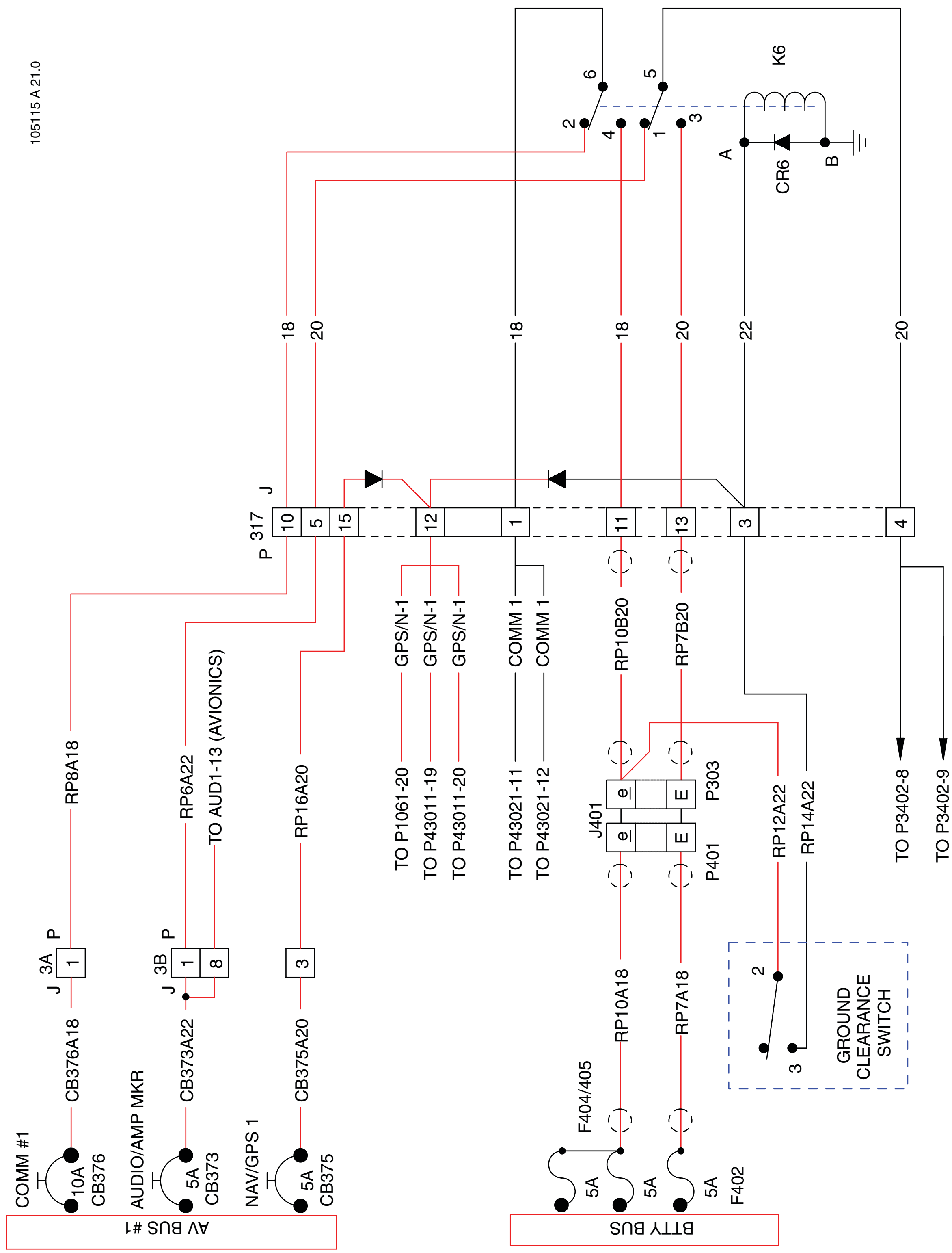
Ground Clearance
Figure 1 (Sheet 2 of 6)



Ground Clearance
Figure 1 (Sheet 3 of 6)

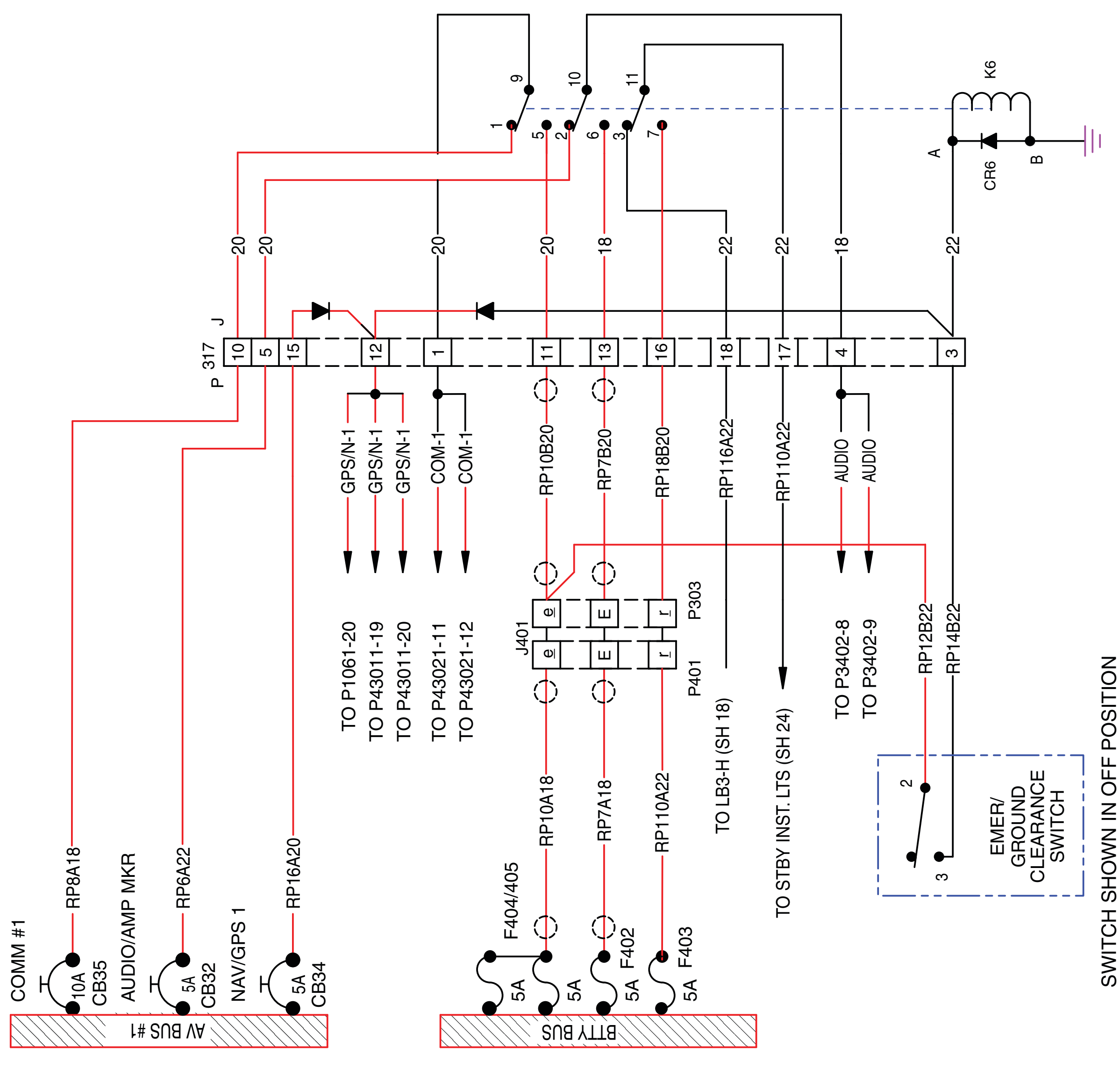


Ground Clearance
Figure 1 (Sheet 4 of 6)



Ground Clearance
Figure 1 (Sheet 5 of 6)

105805 New 25.0



Ground Clearance
Figure 1 (Sheet 6 of 6)

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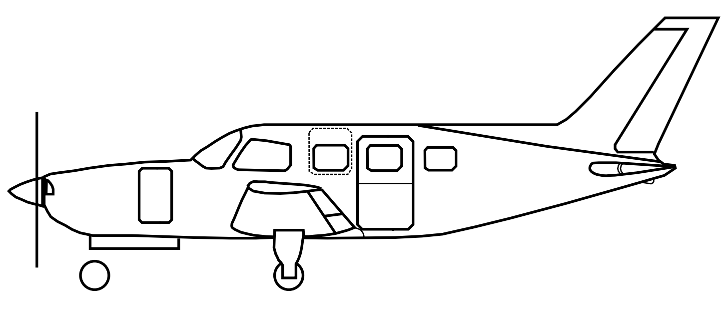
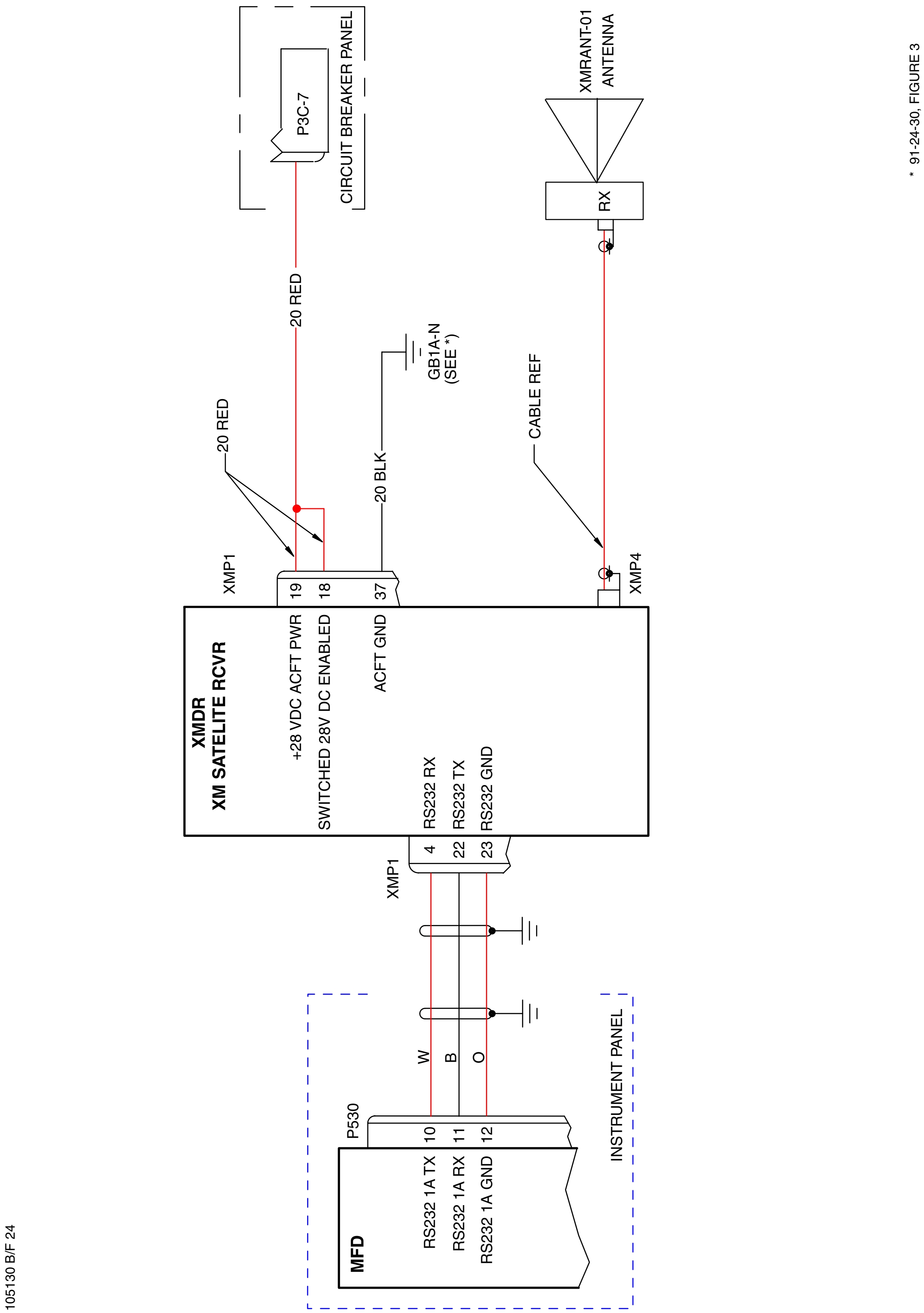
Item #	Designation	Description

Information Pending - See Parts Catalog



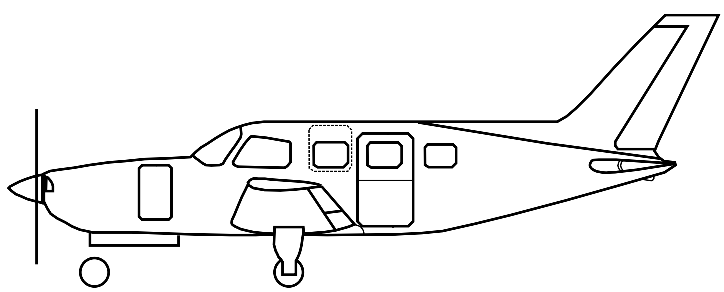
XM Receiver
Figure 1 (Sheet 1 of 2)



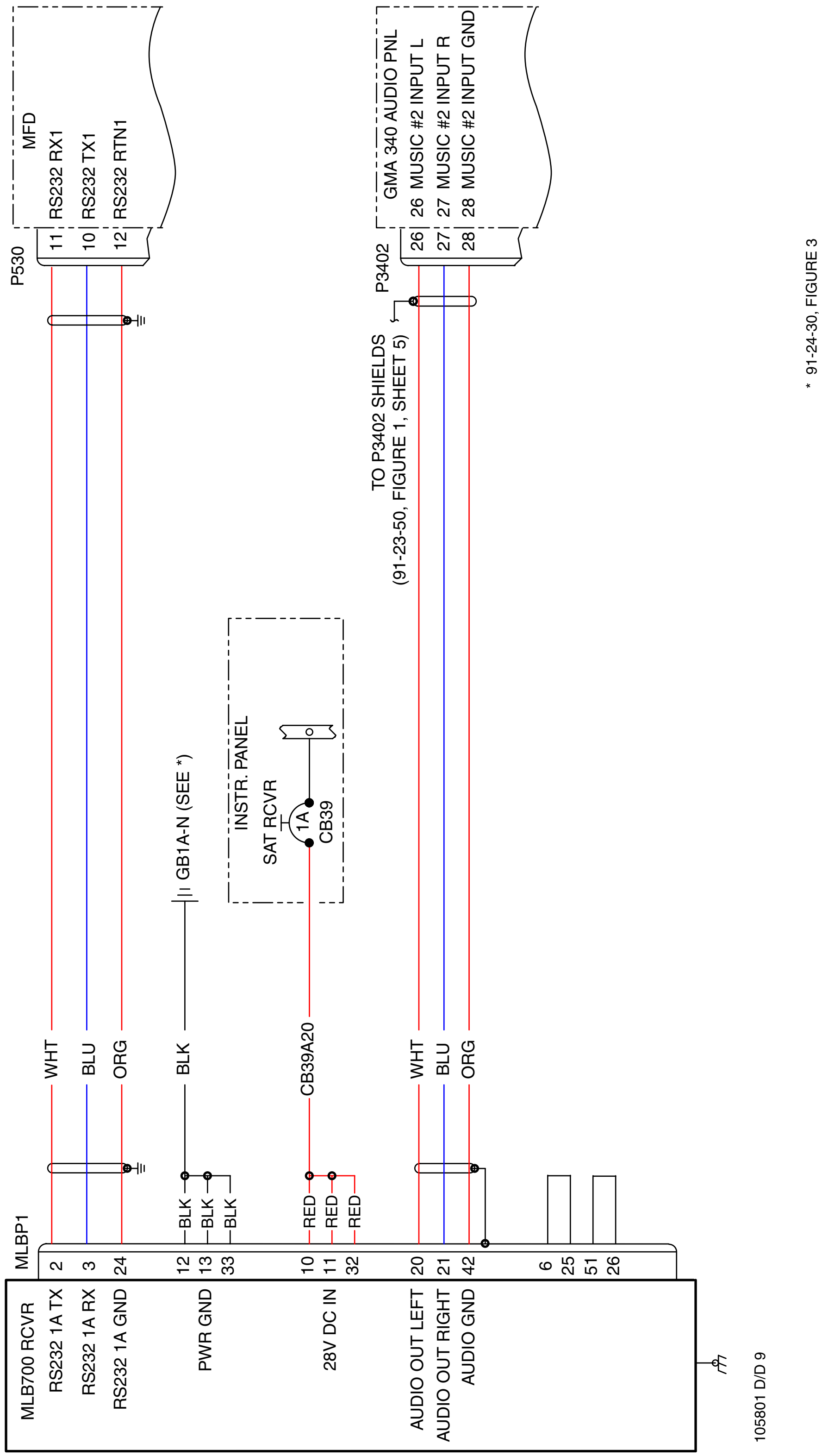
XM Receiver
Figure 1 (Sheet 2 of 2)

Item #	Designation	Description

Information Pending - See Parts Catalog



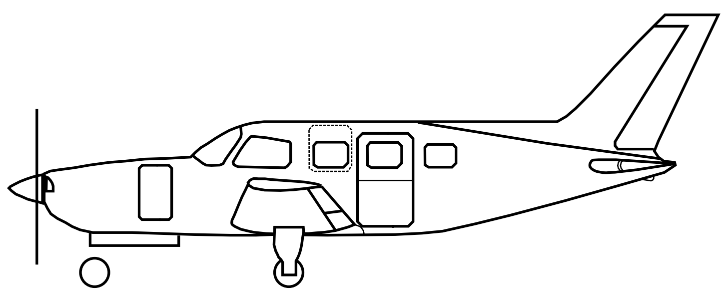
SAT Receiver
Figure 2 (Sheet 1 of 2)



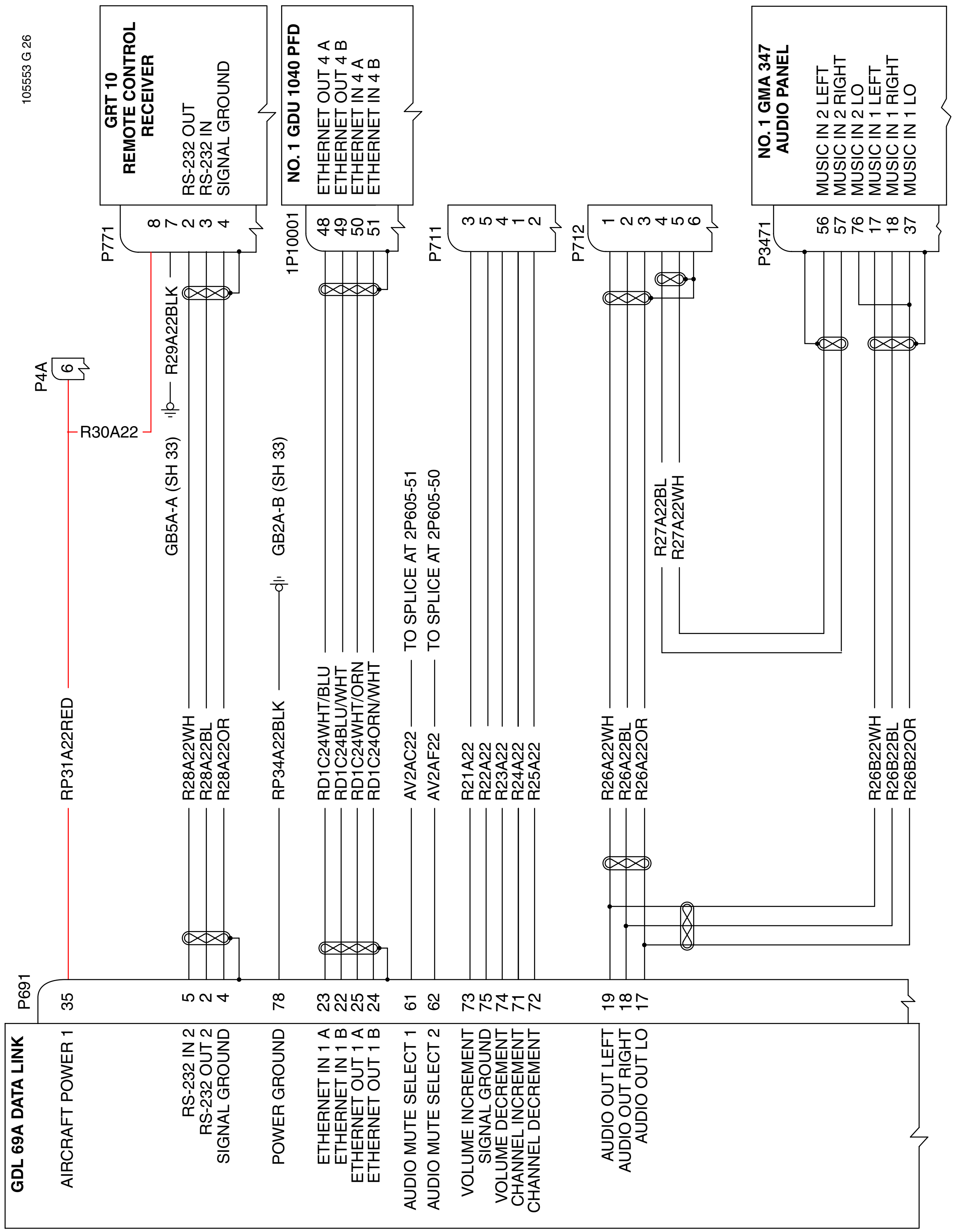
SAT Receiver
Figure 2 (Sheet 2 of 2)

Item #	Designation	Description

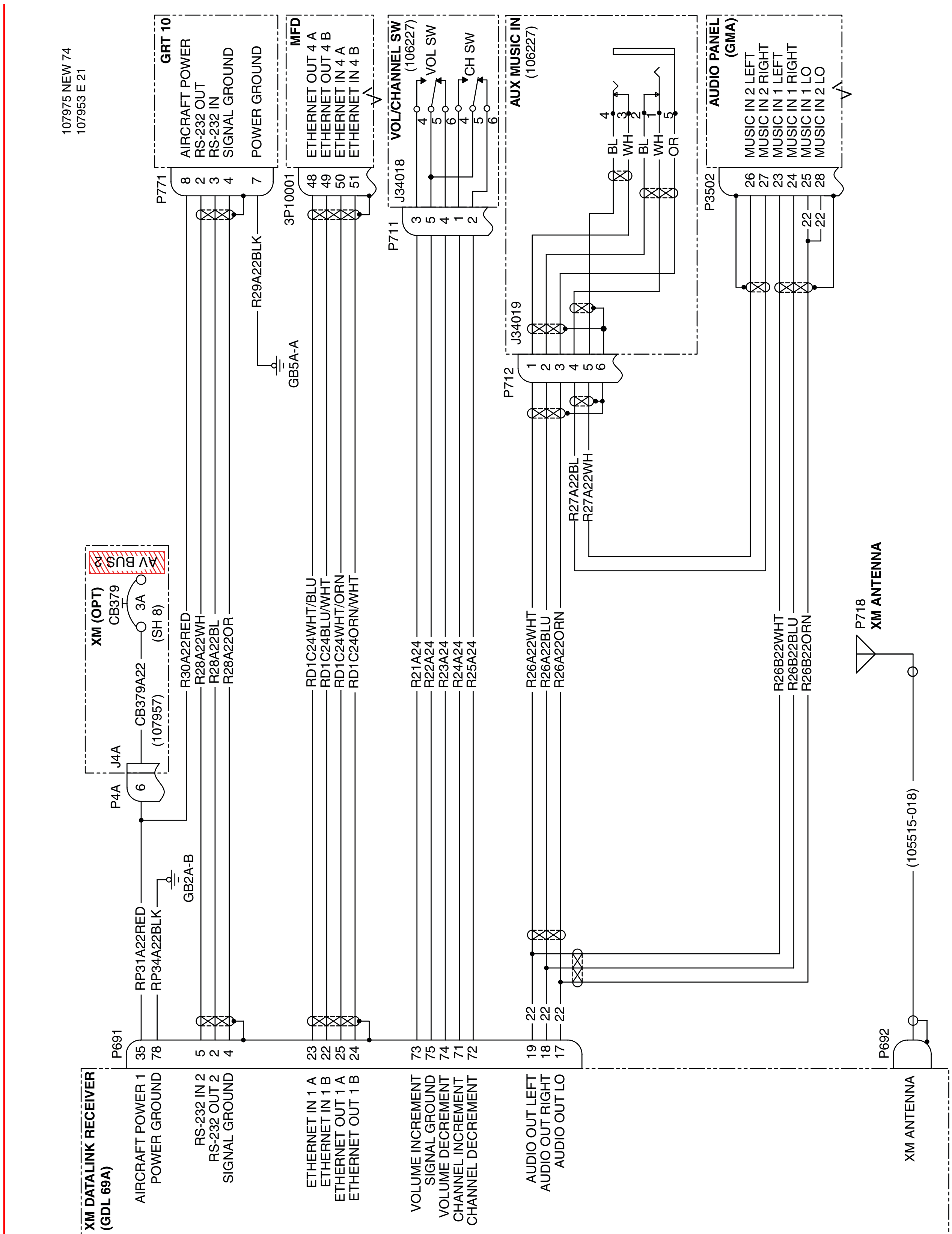
Information Pending - See Parts Catalog



IAS - G1000 - Data Link (GDL-69A)
Figure 3 (Sheet 1 of 3)

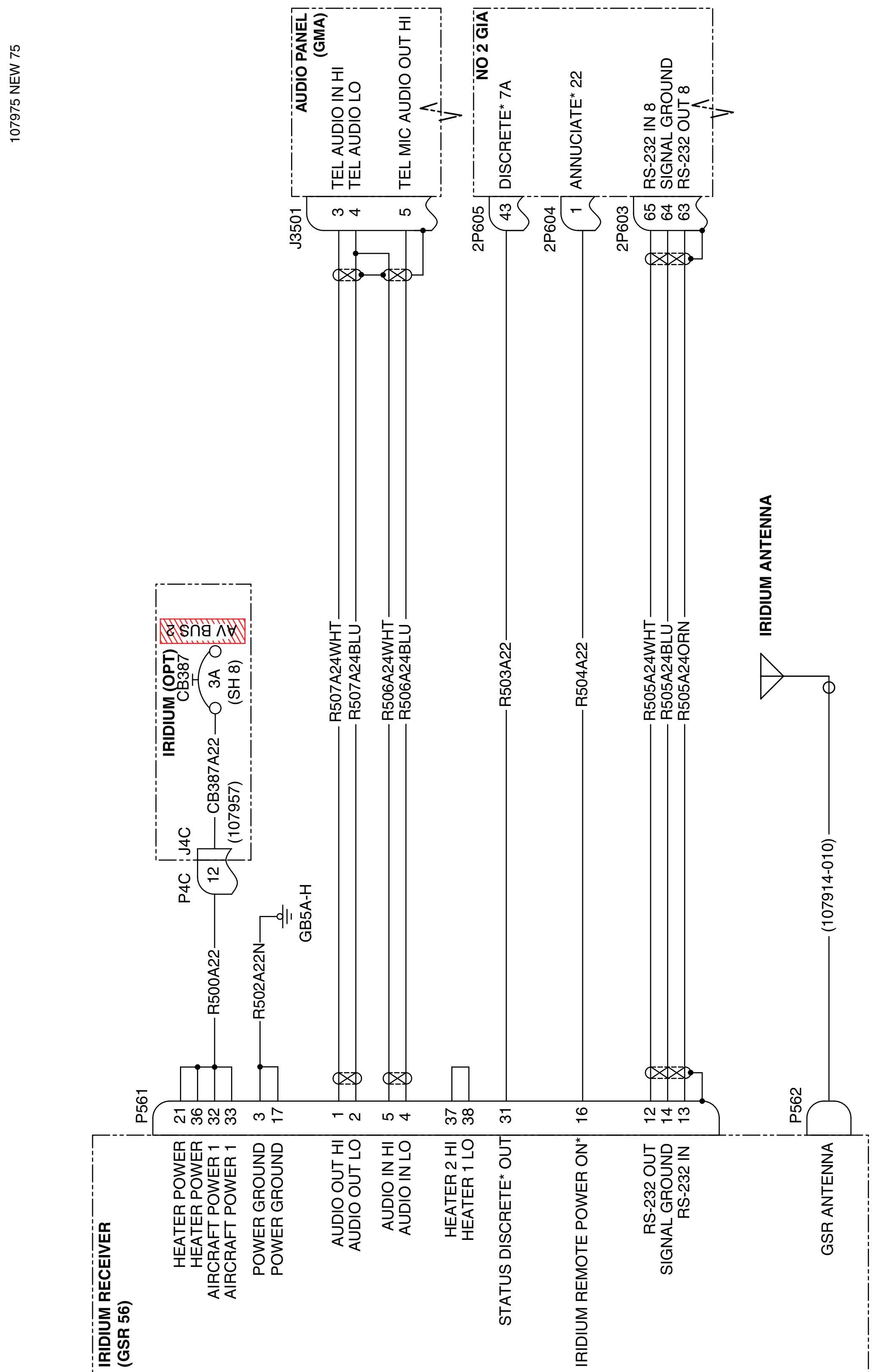


IAS - G1000 - Data Link (GDL-69A)
Figure 3 (Sheet 2 of 3)



IAS - G1000 - Data Link (GDL-69A)

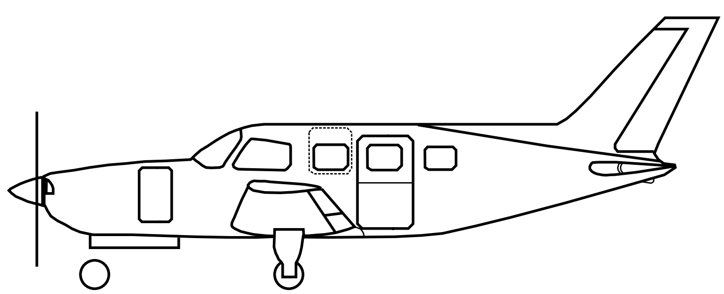
Figure 3 (Sheet 3 of 3)



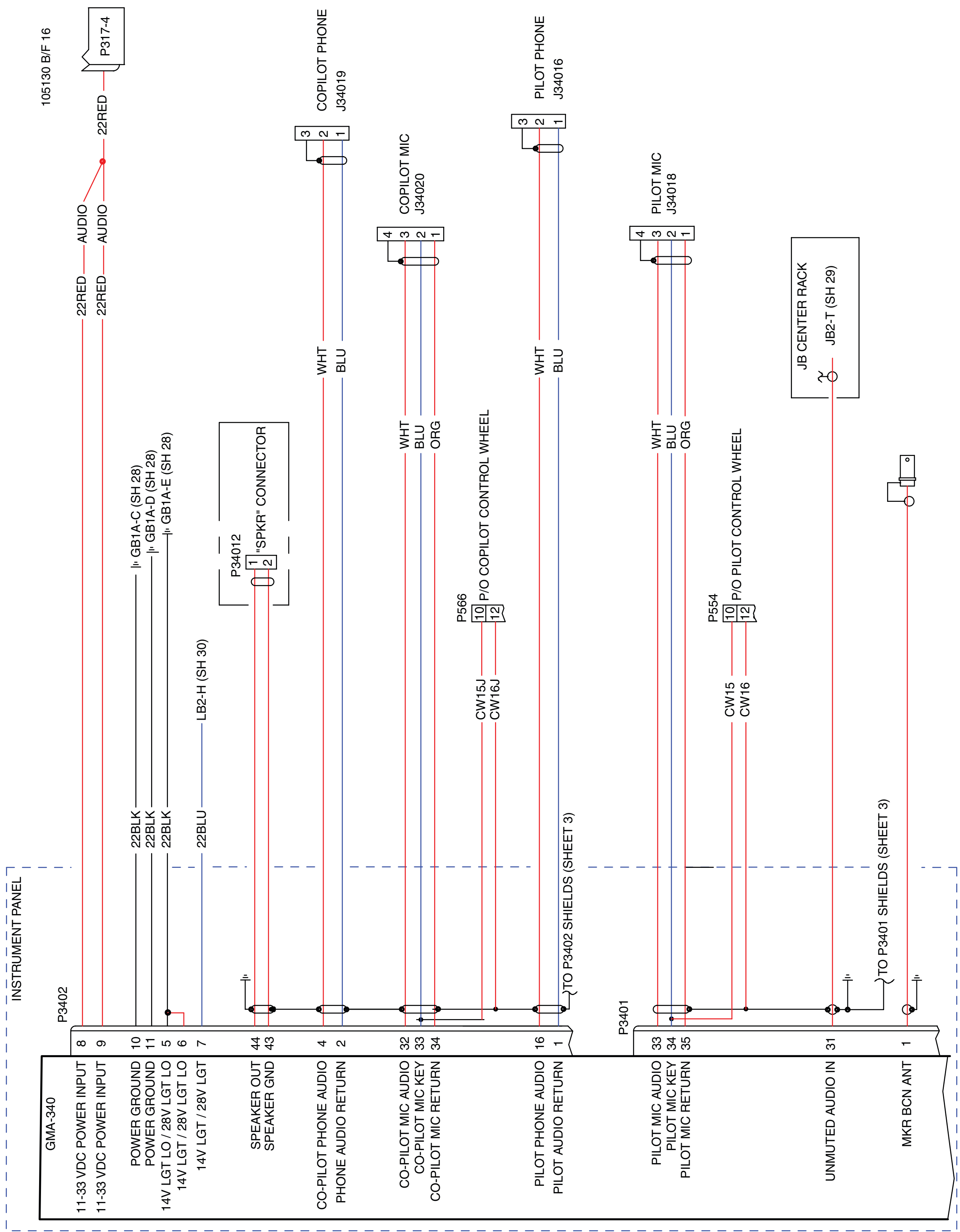
Iridium Satellite Receiver GSR 56 (Optional)
Figure 4

Item #	Designation	Description

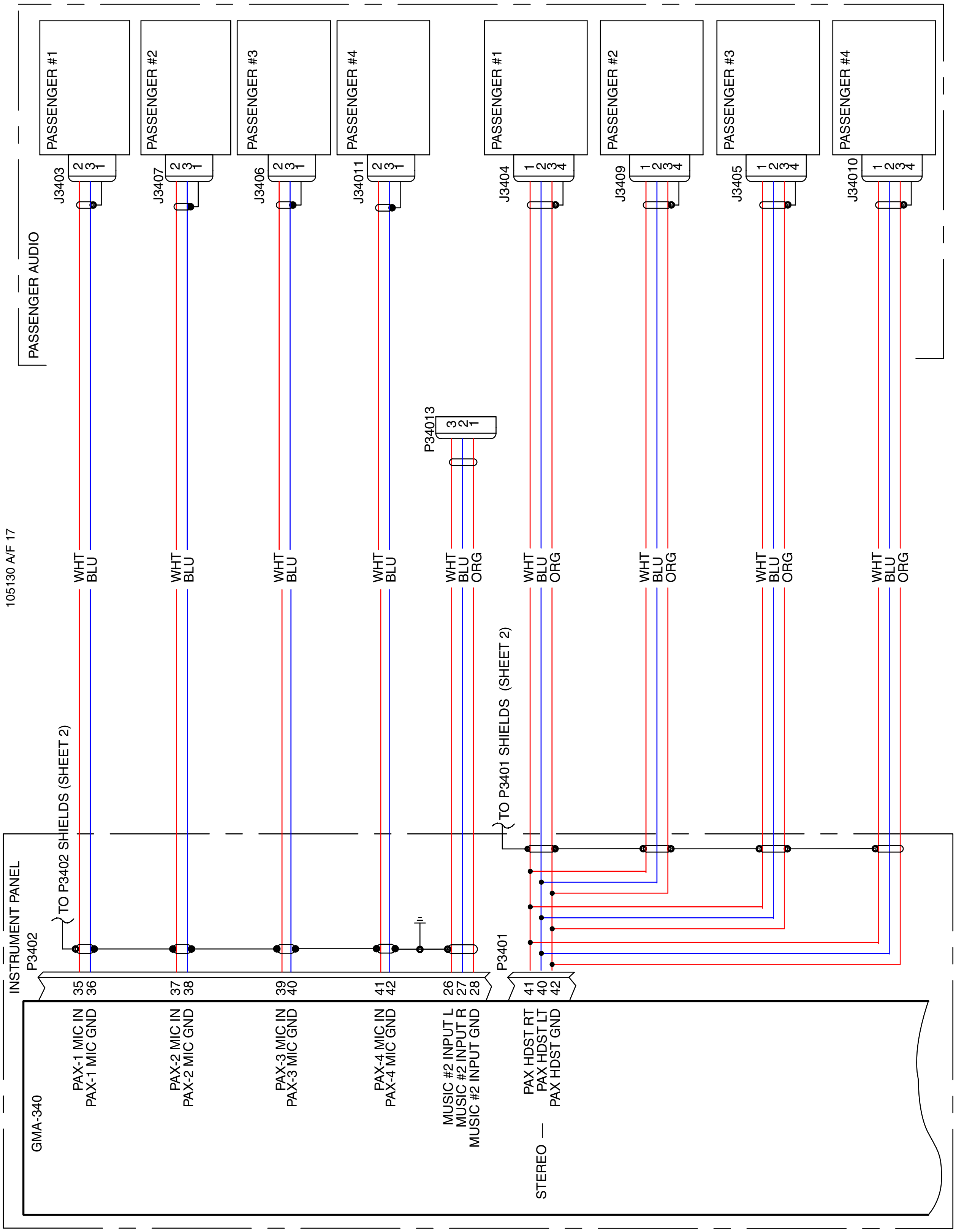
Information Pending - See Parts Catalog



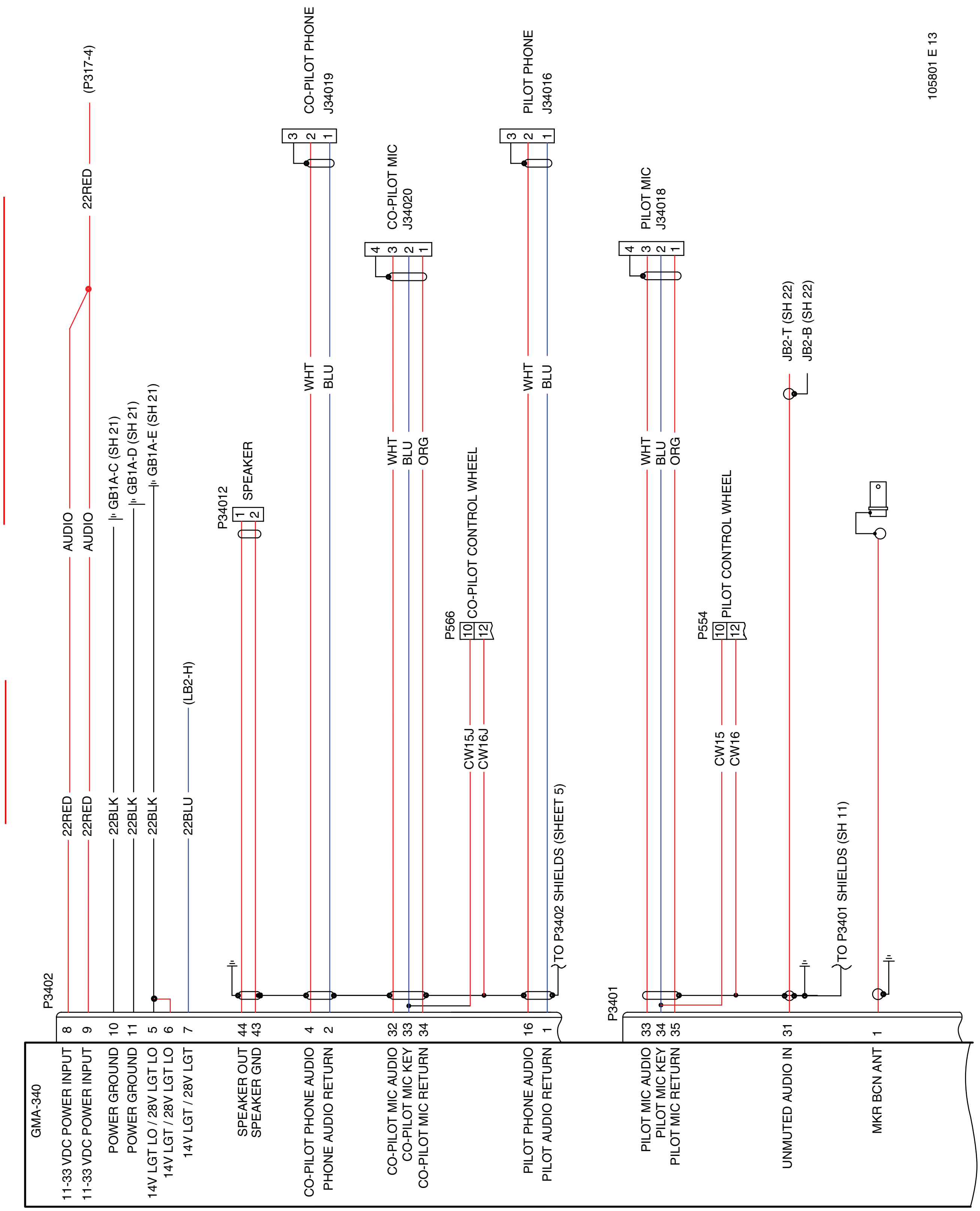
GMA 340 Audio/Mkr/Intercom
Figure 1 (Sheet 1 of 5)



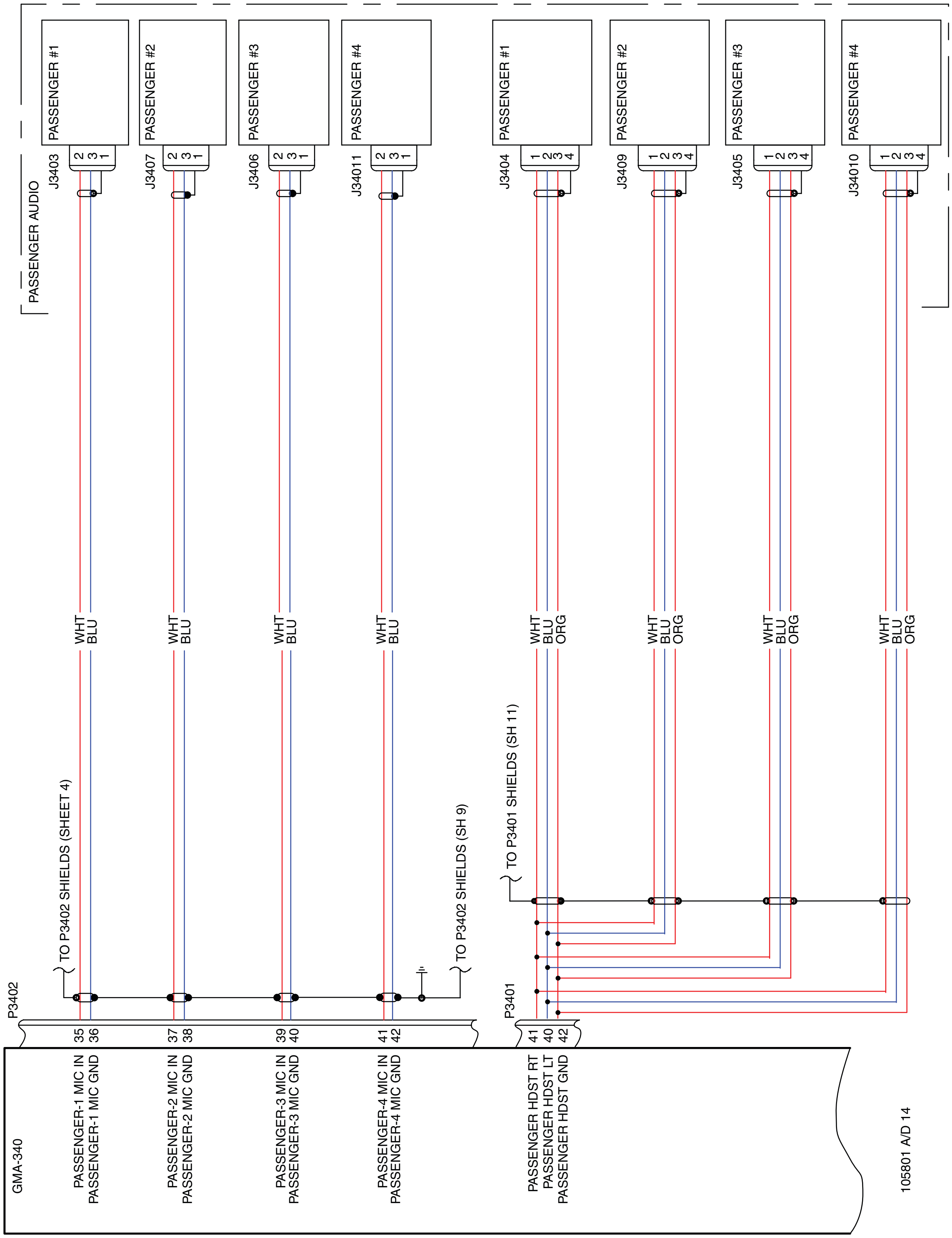
GMA 340 Audio/Mkr/Intercom
Figure 1 (Sheet 2 of 5)



GMA 340 Audio/Mkr/Intercom
Figure 1 (Sheet 3 of 5)



GMA 340 Audio/Mkr/Intercom
Figure 1 (Sheet 4 of 5)



GMA 340 Audio/Mkr/Intercom
Figure 1 (Sheet 5 of 5)

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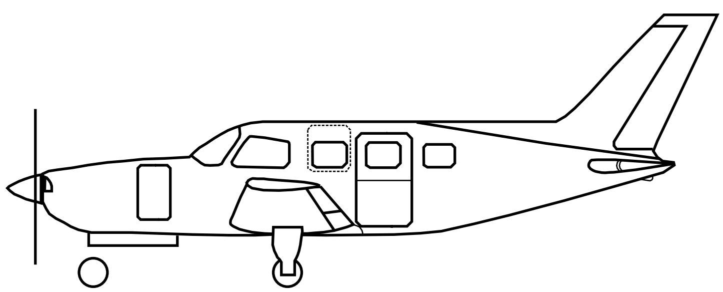
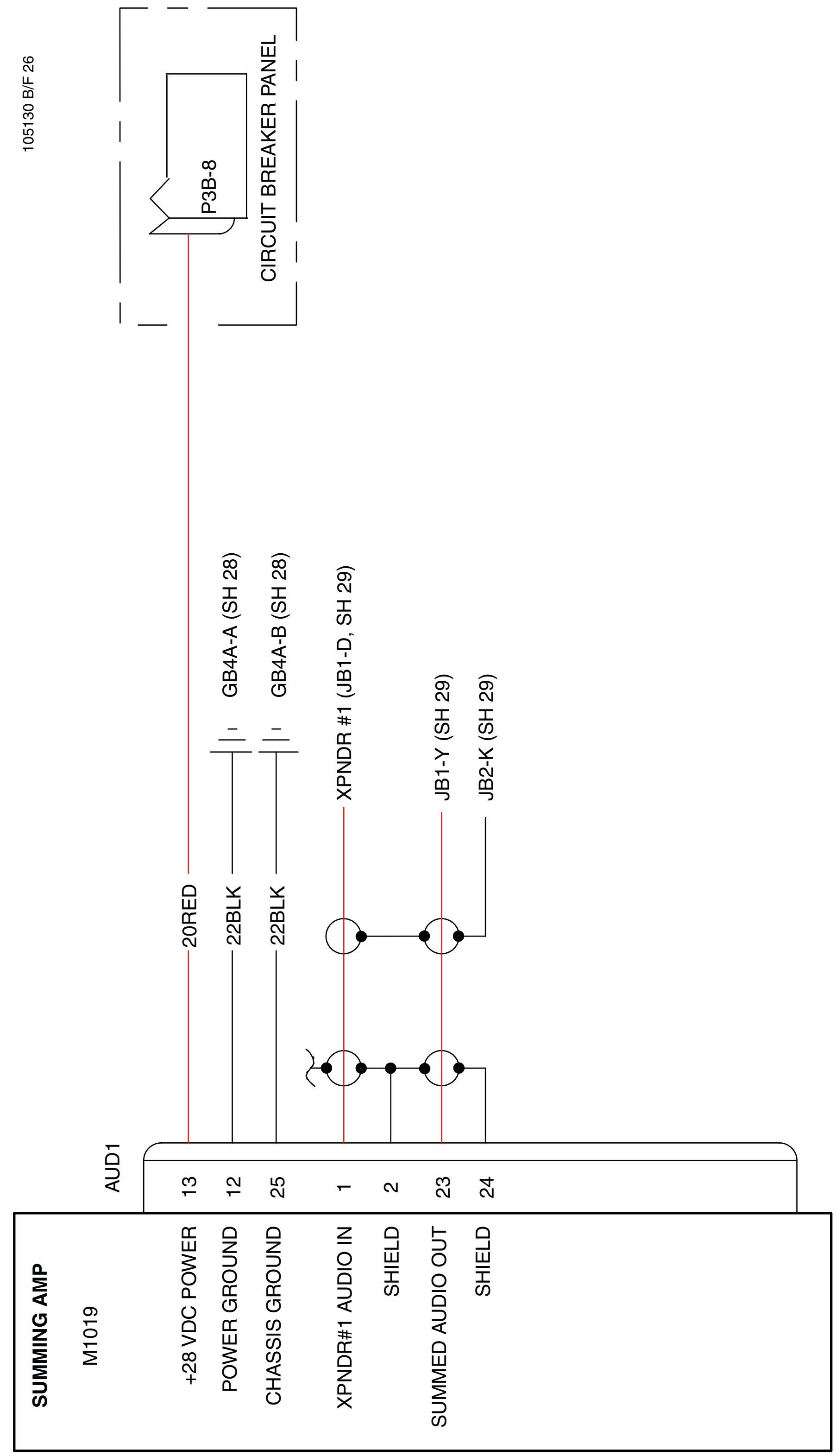
Item #	Designation	Description

Information Pending - See Parts Catalog



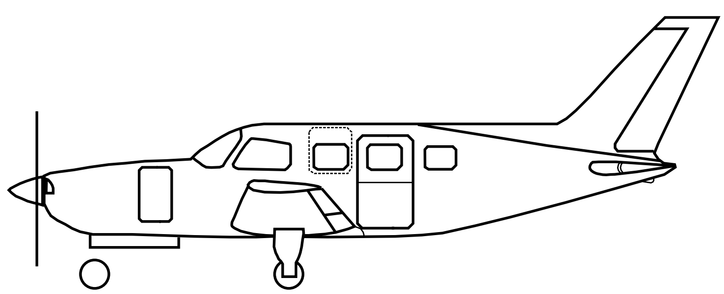
Summing Amp
Figure 2 (Sheet 1 of 2)



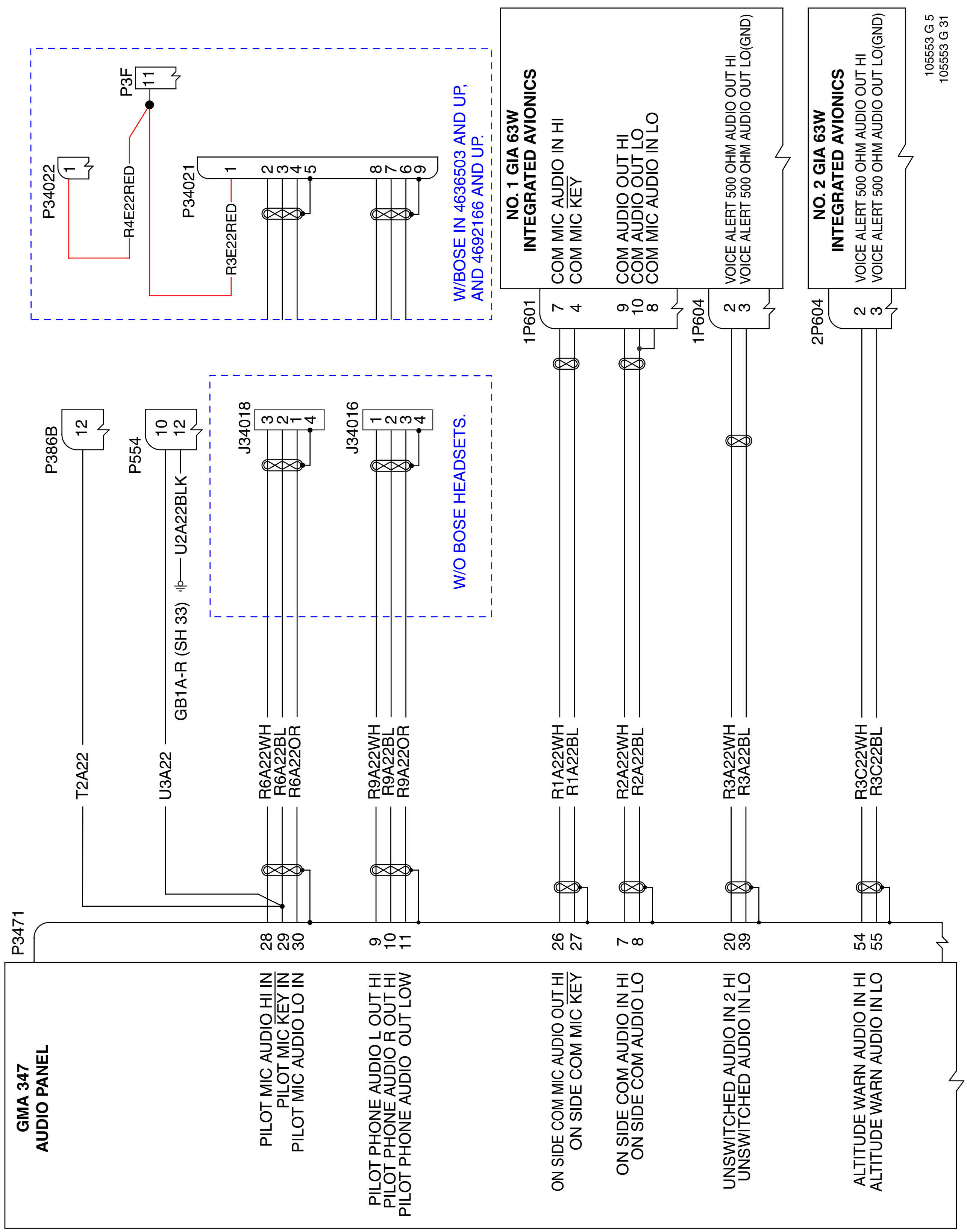
Summing Amp
Figure 2 (Sheet 2 of 2)

Item #	Designation	Description

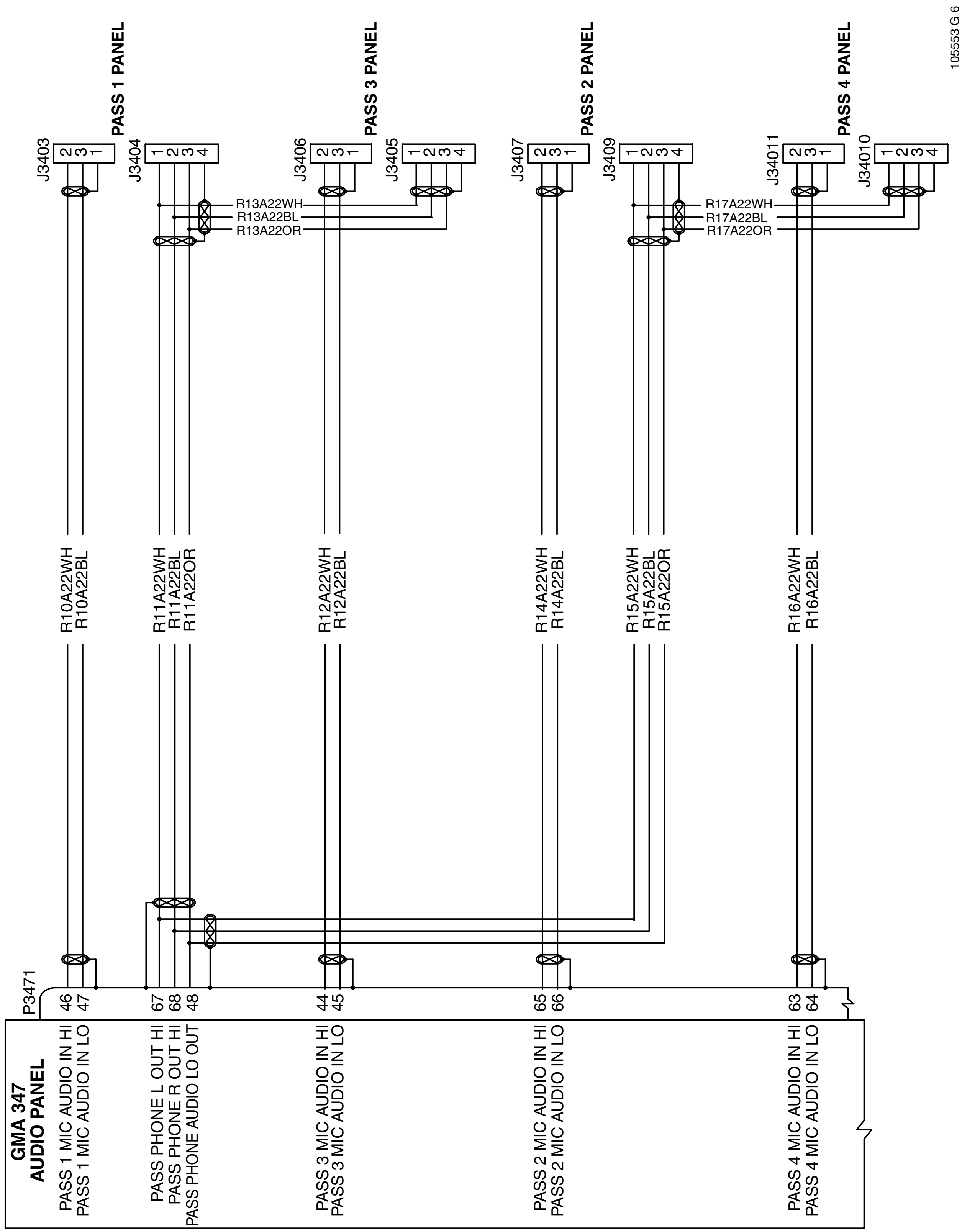
Information Pending - See Parts Catalog



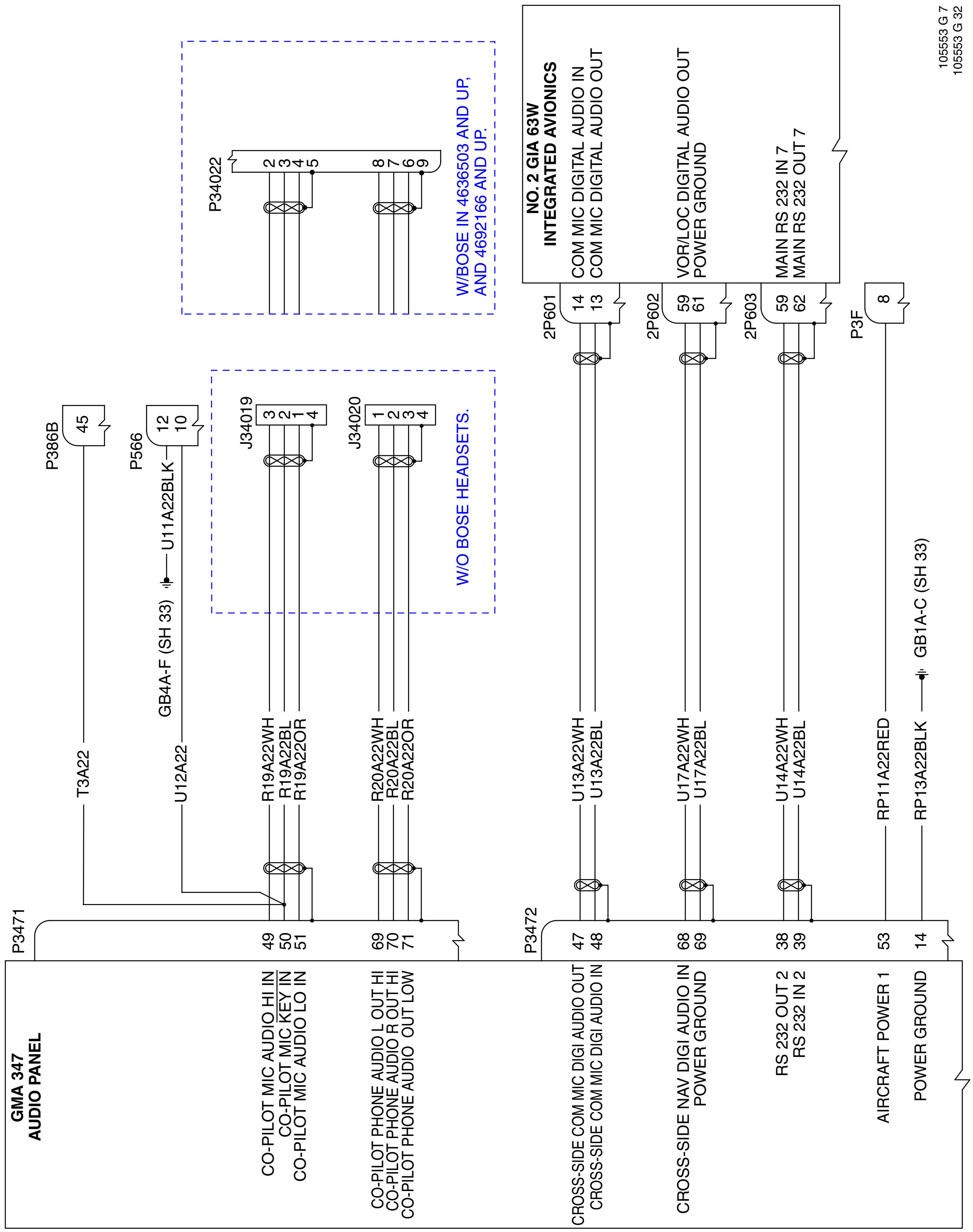
GMA 347 Audio/Mkr/Intercom
Figure 3 (Sheet 1 of 5)



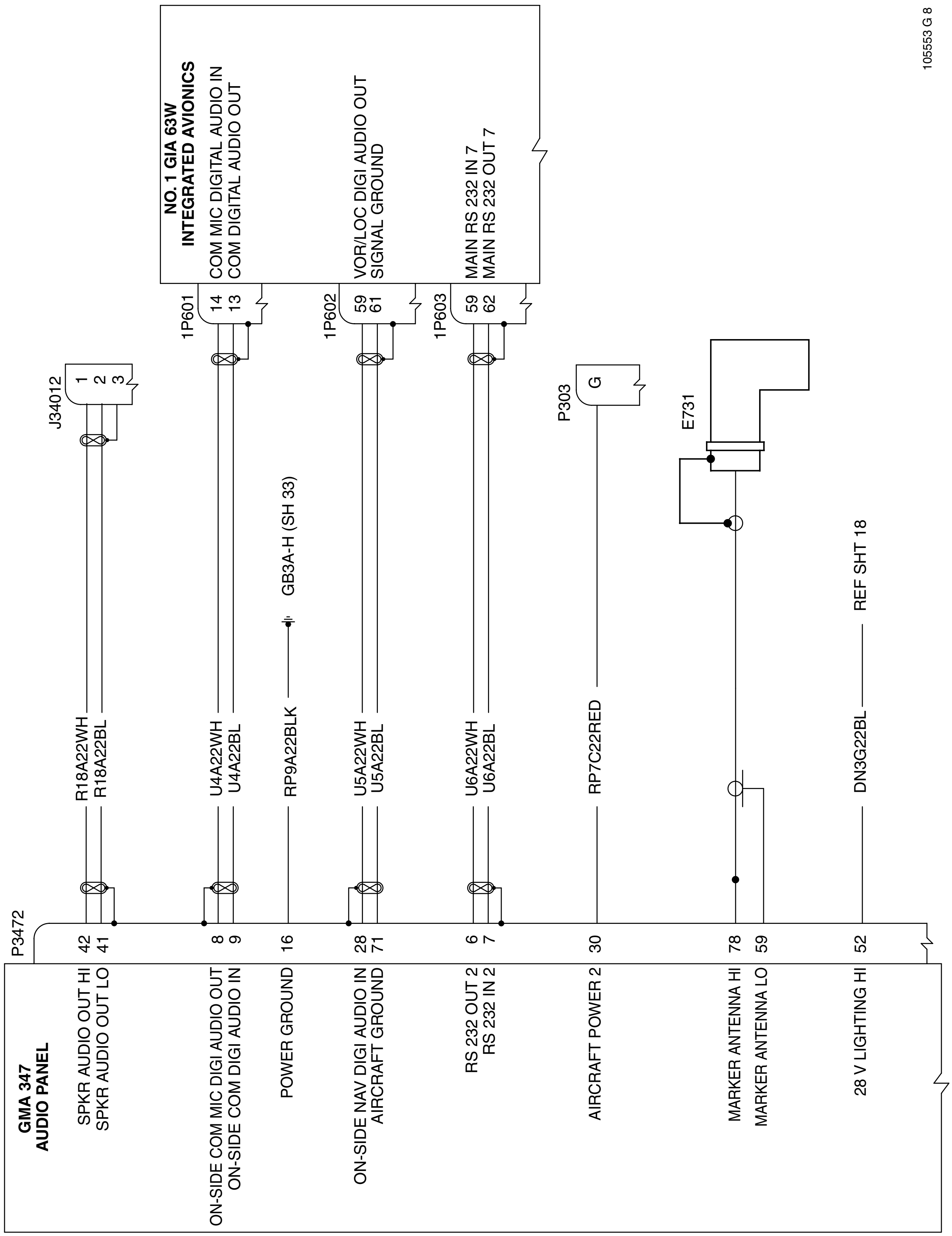
GMA 347 Audio/Mkr/Intercom
Figure 3 (Sheet 2 of 5)



GMA 347 Audio/Mkr/Intercom
Figure 3 (Sheet 3 of 5)



GMA 347 Audio/Mkr/Intercom
Figure 3 (Sheet 4 of 5)



GMA 347 Audio/Mkr/Intercom
Figure 3 (Sheet 5 of 5)

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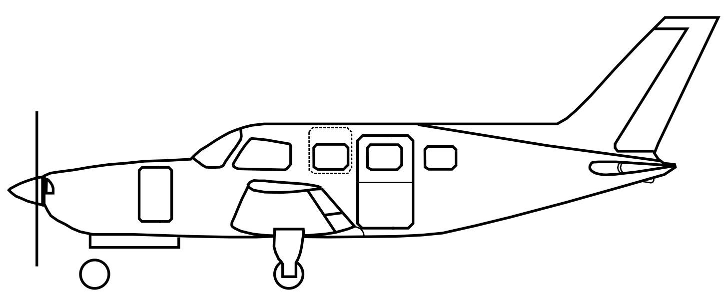
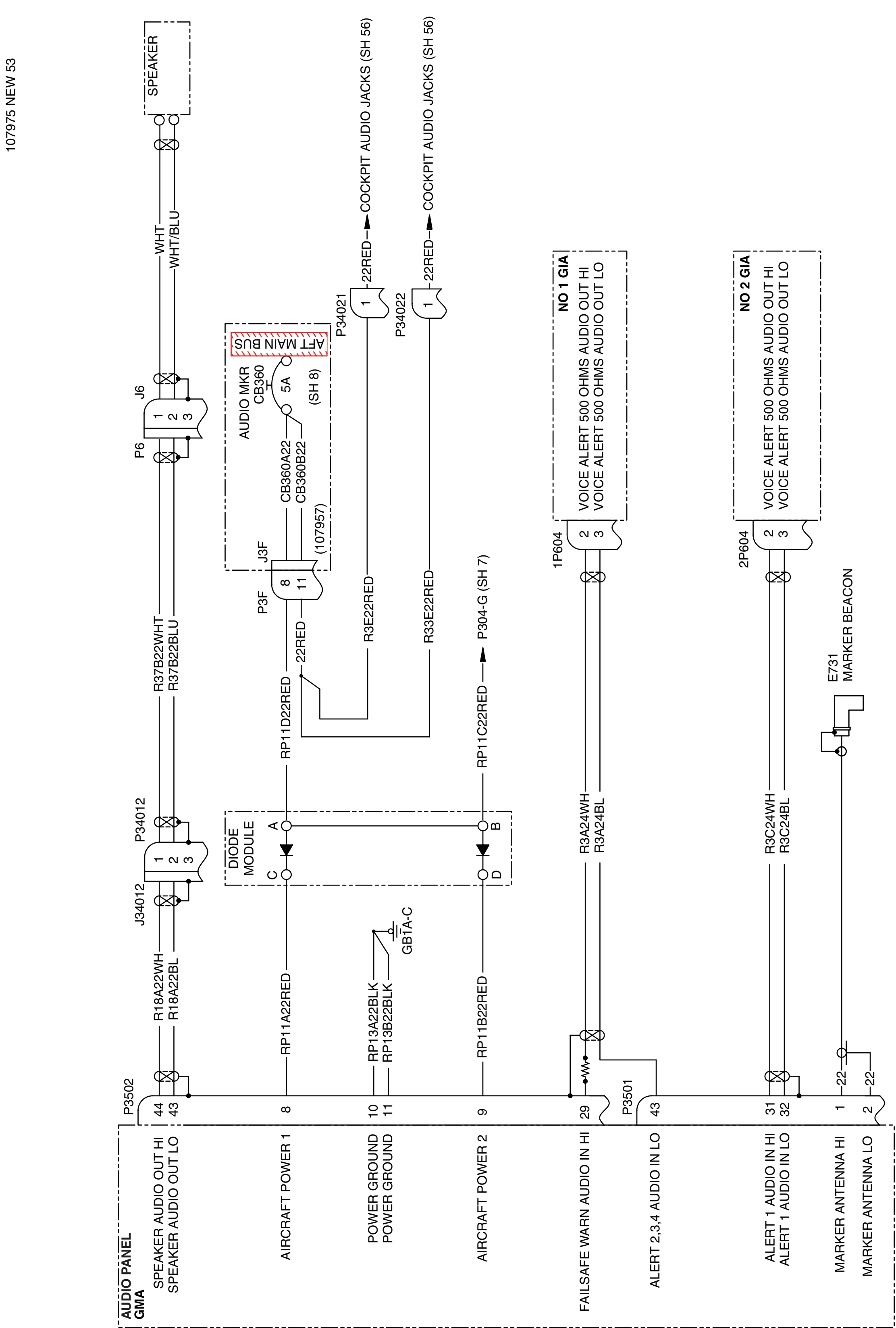
Item #	Designation	Description

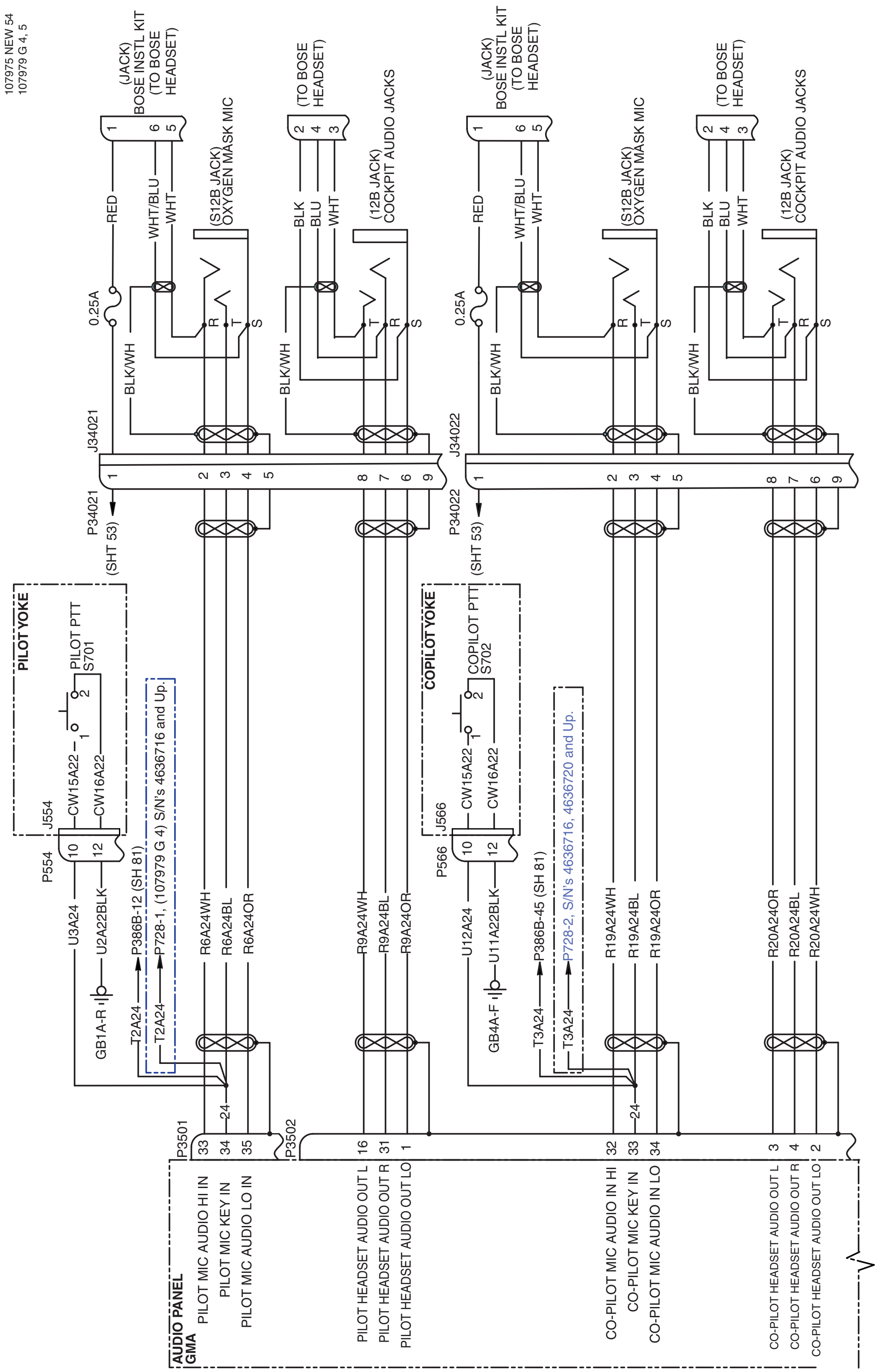
Information Pending - See Parts Catalog



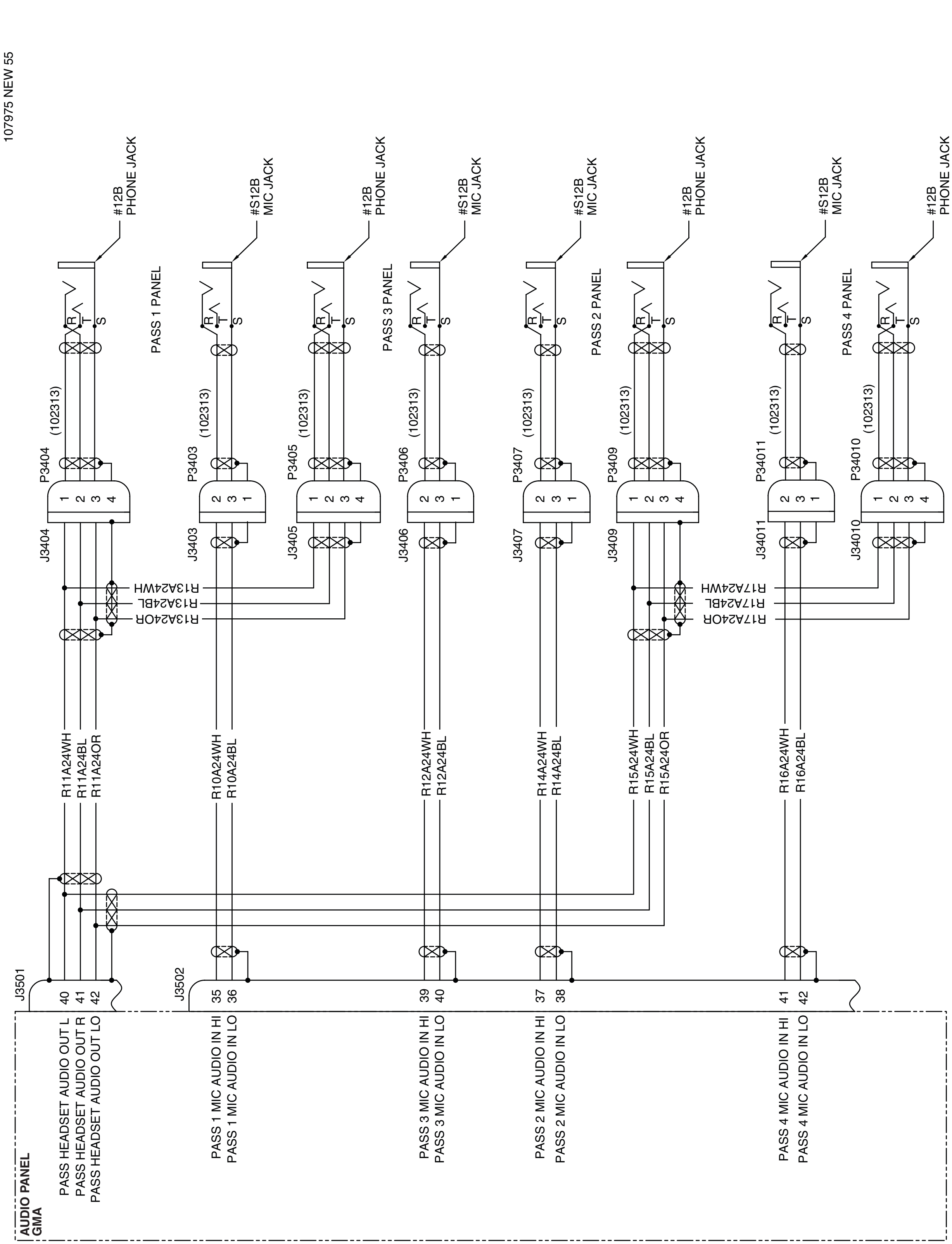
GMA 350 Audio/Marker/Intercom
Figure 4 (Sheet 1 of 4)



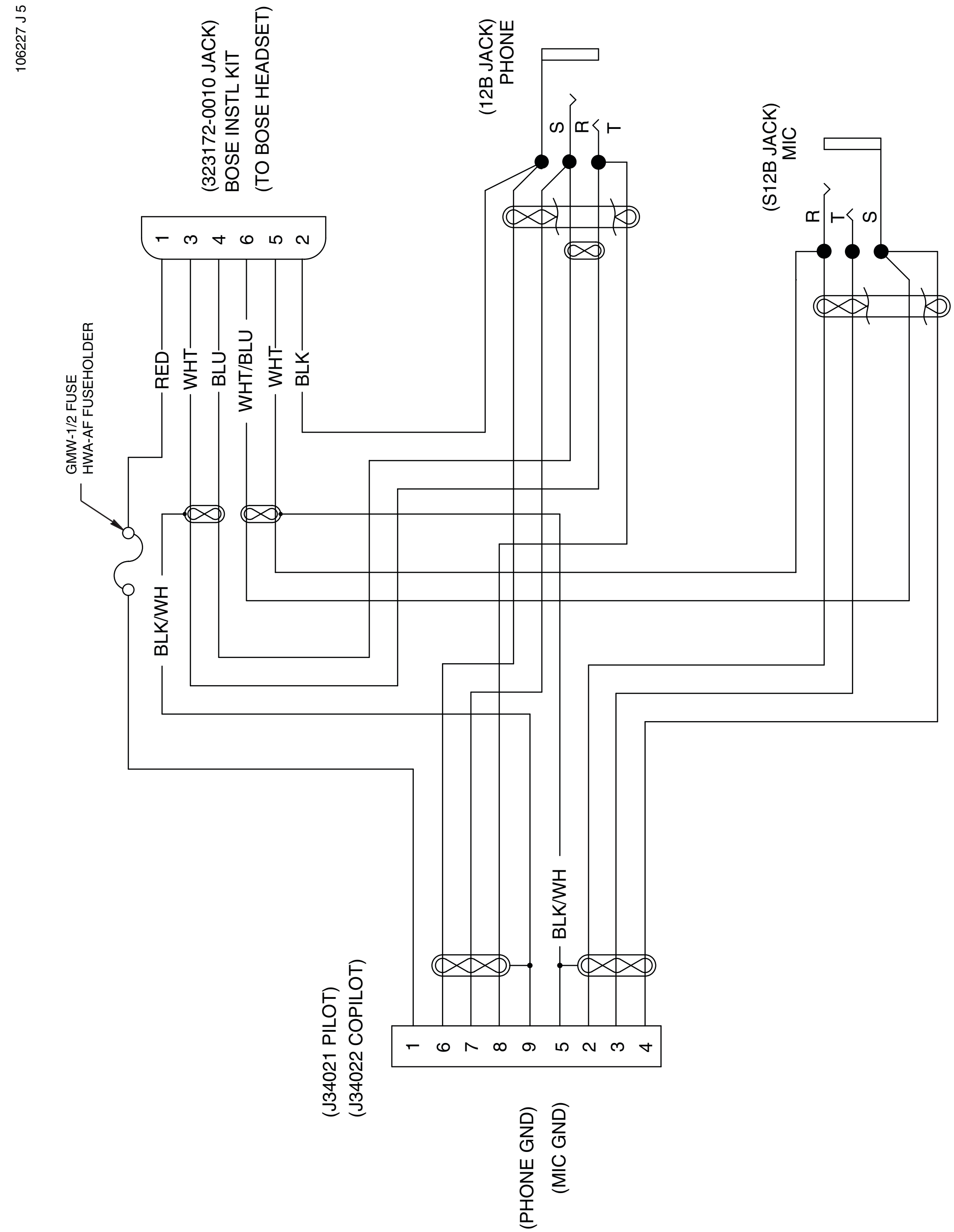
GMA 350 Audio/Marker/Intercom
Figure 4 (Sheet 2 of 4)



GMA 350 Audio/Marker/Intercom
Figure 4 (Sheet 3 of 4)



GMA 350 Audio/Marker/Intercom
Figure 4 (Sheet 4 of 4)



Bose Headset Option
Figure 5

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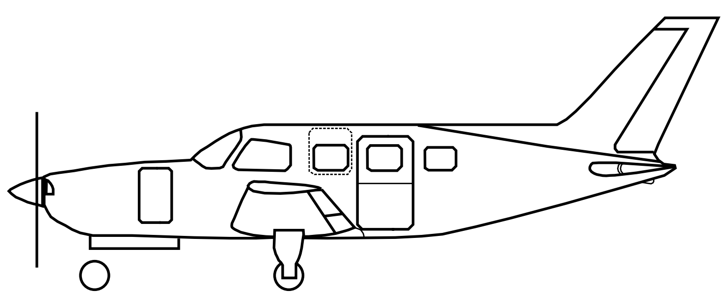
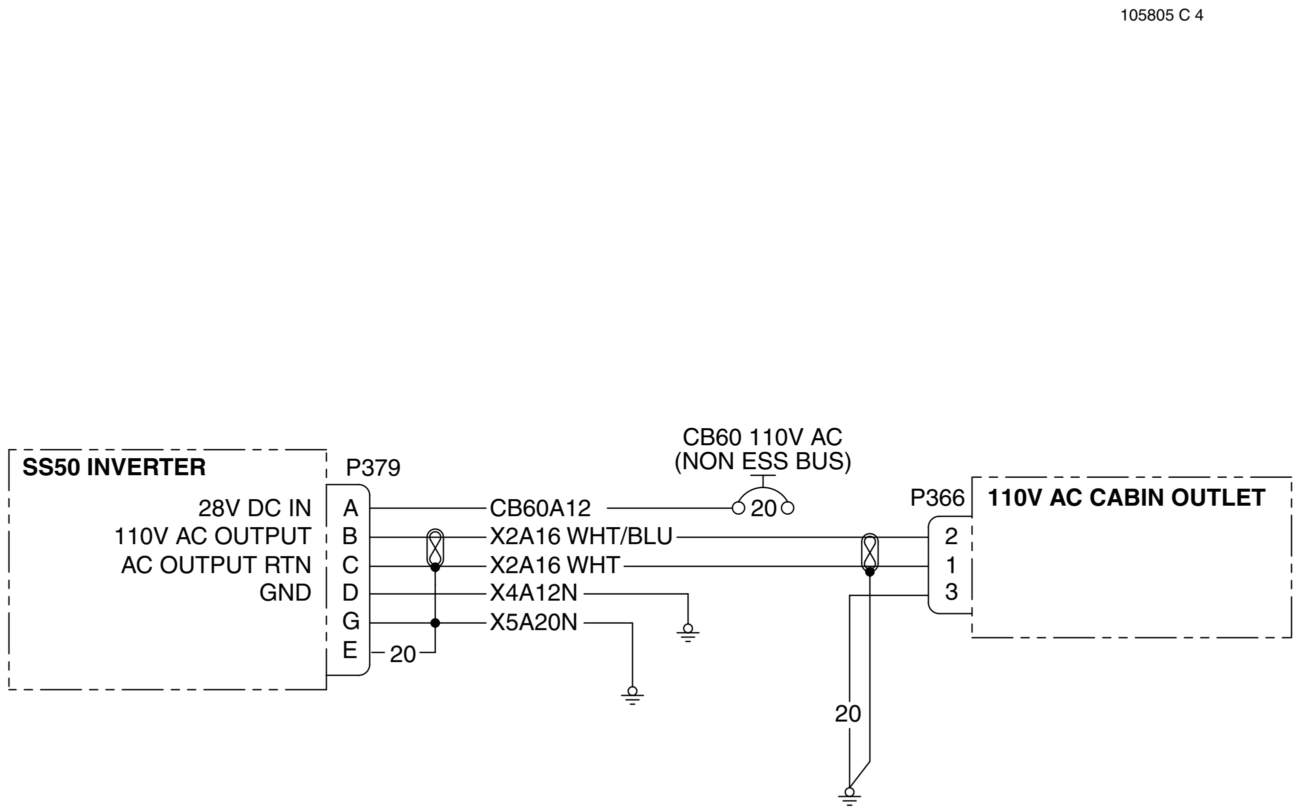
Item #	Designation	Description

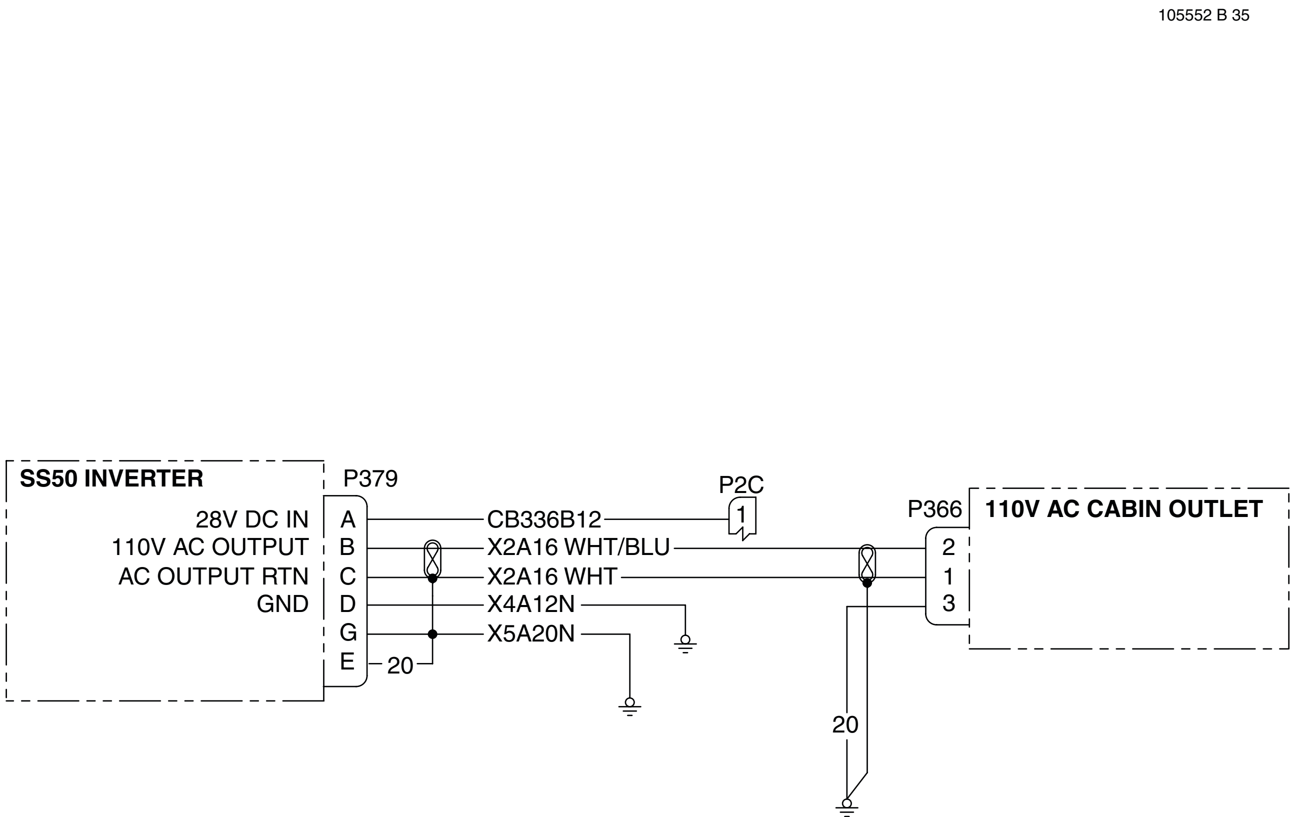
Information Pending - See Parts Catalog



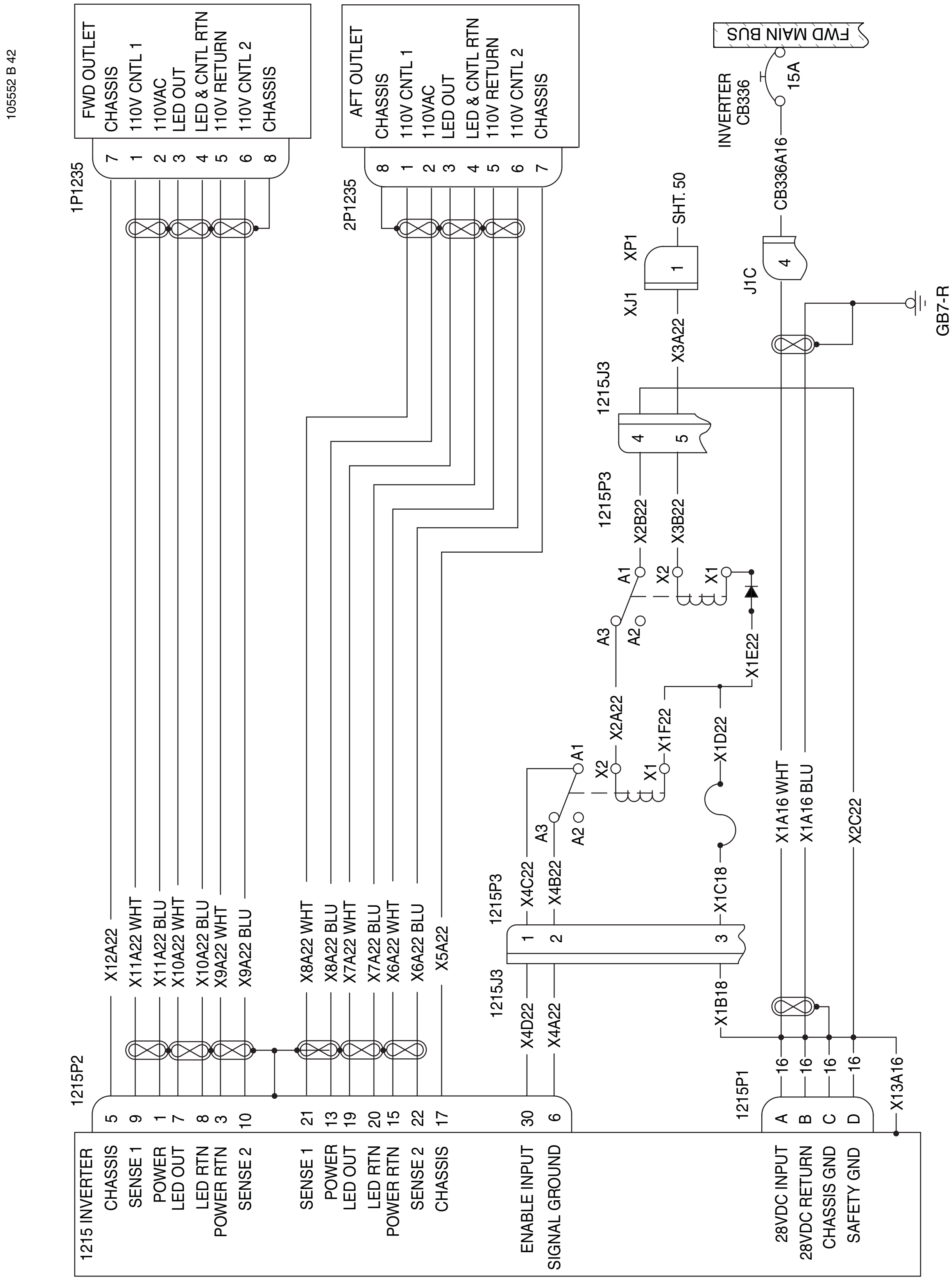
VAC Inverter
Figure 1 (Sheet 1 of 7)



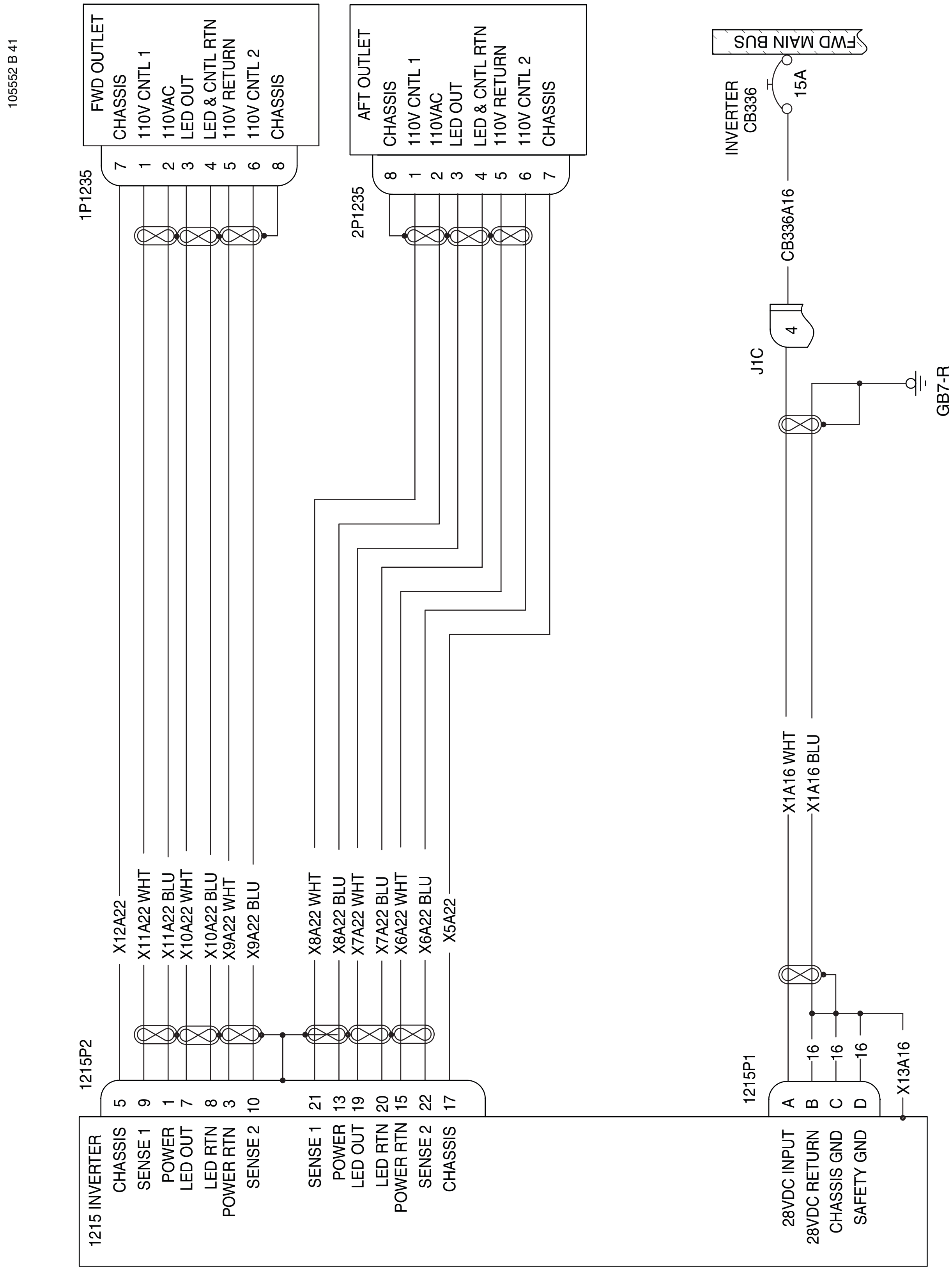
VAC Inverter
Figure 1 (Sheet 2 of 7)



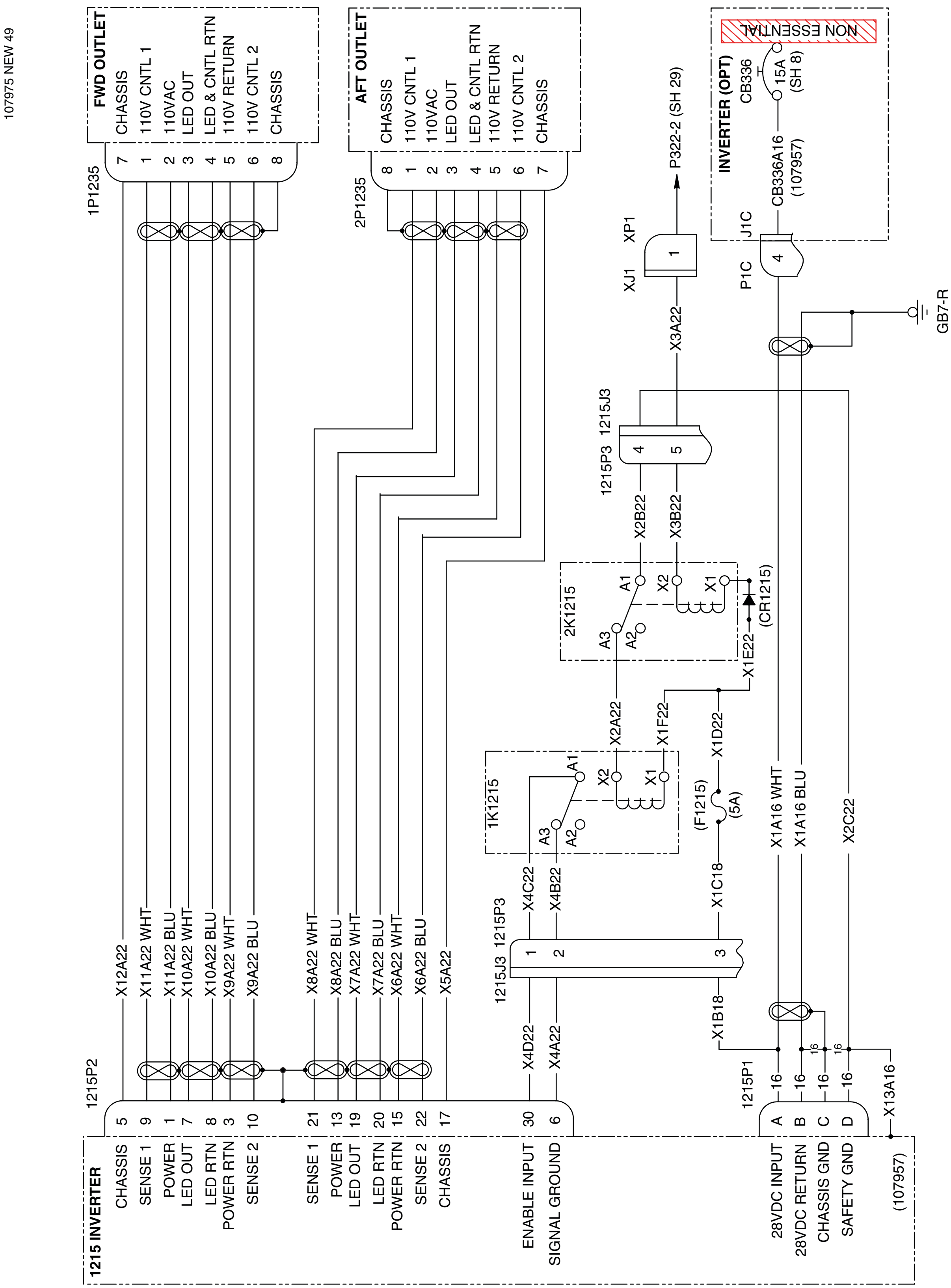
VAC Inverter
Figure 1 (Sheet 3 of 7)



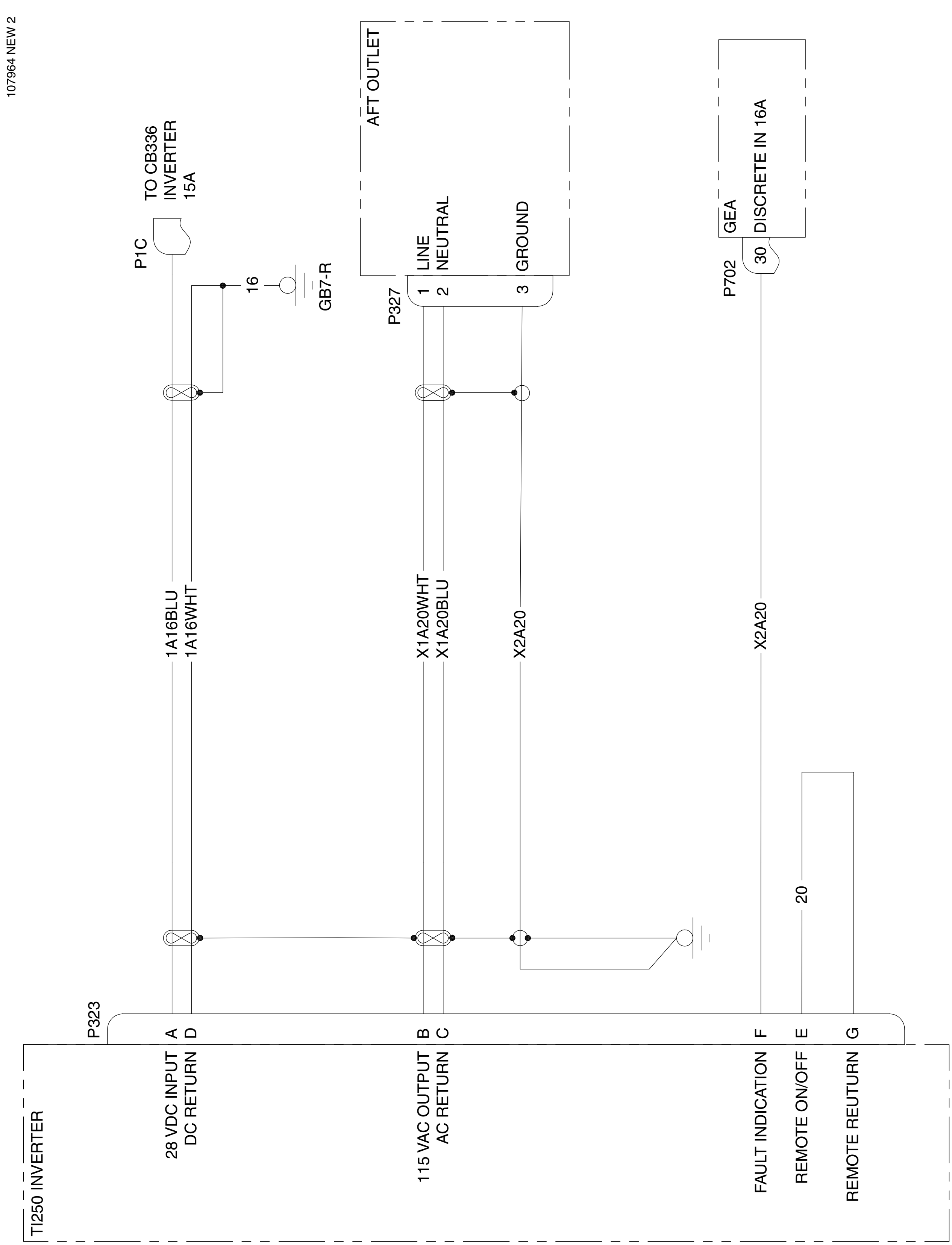
VAC Inverter
Figure 1 (Sheet 4 of 7)



VAC Inverter
Figure 1 (Sheet 5 of 7)



VAC Inverter
Figure 1 (Sheet 6 of 7)



VAC Inverter
Figure 1 (Sheet 7 of 7)

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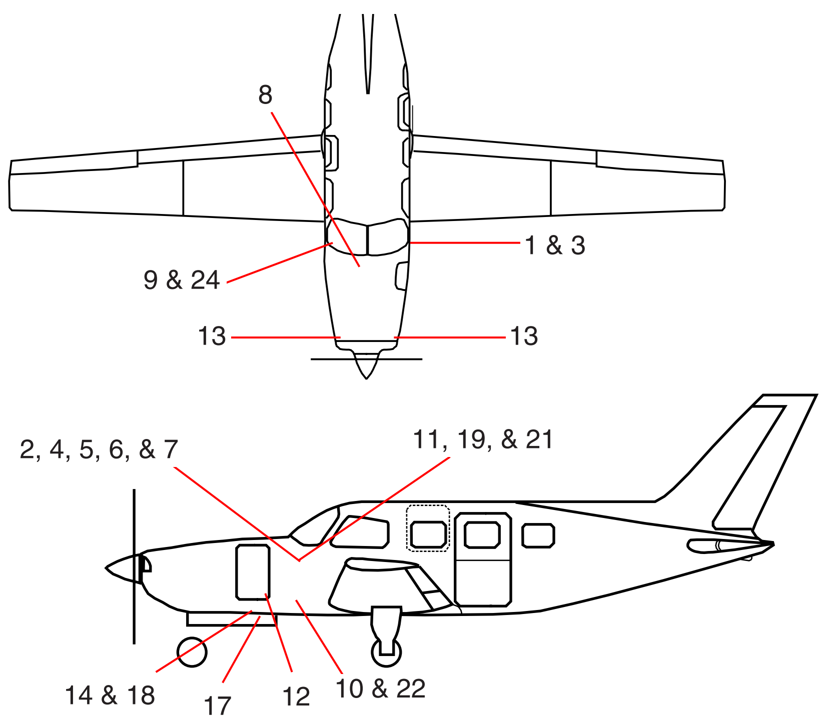
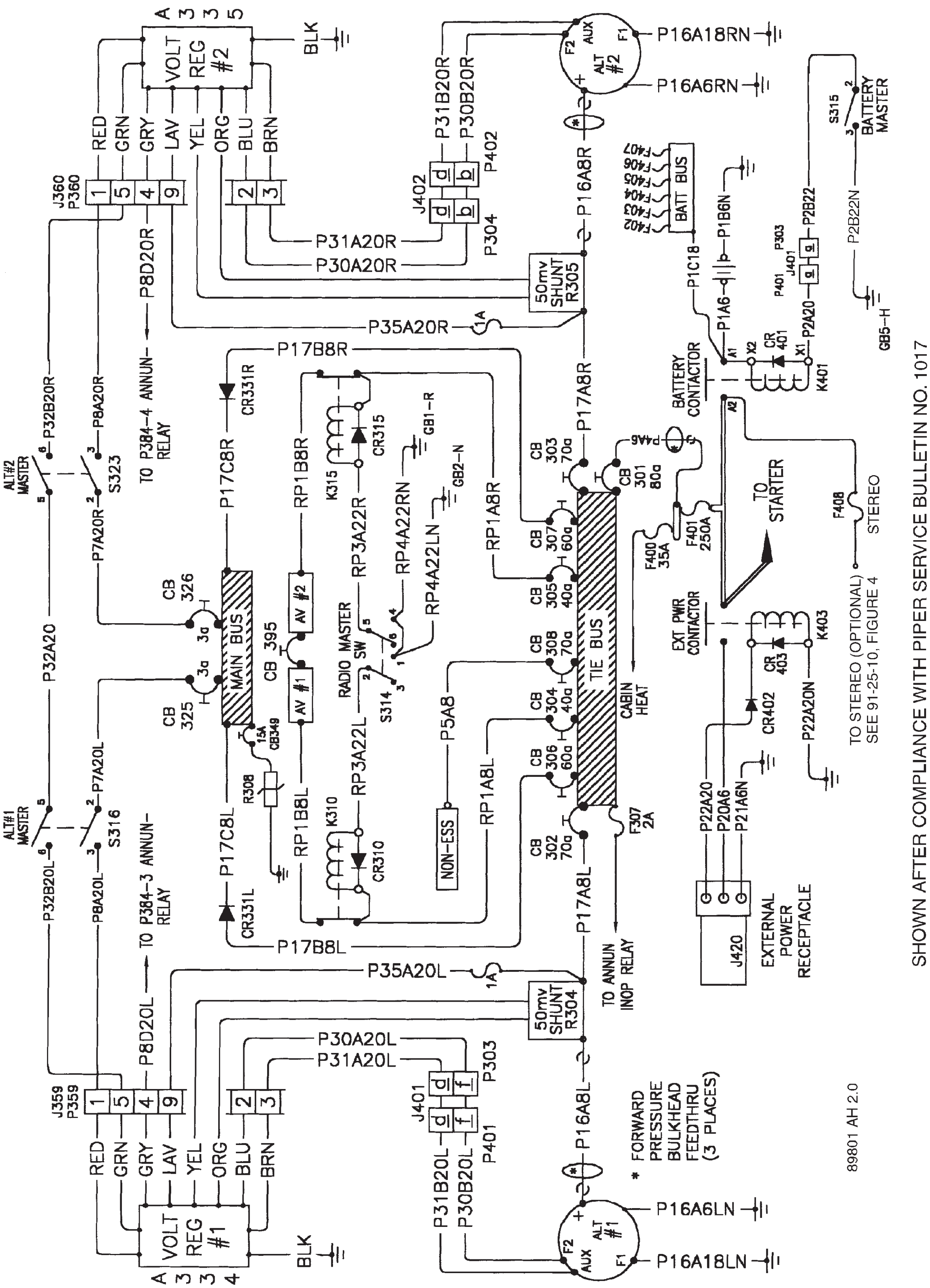
Item #	Designation	Description
1	CB325, CB326, CB9, CB10	Circuit Breaker (3 Amp or 5 Amp)
2	CB306, CB307, CB64, CB65	Circuit Breaker (60 Amp or 80 Amp)
3	CB349, CB31	Circuit Breaker (15 Amp)
4	CB302, CB303, CB308, CB62, CB63, CB69	Circuit Breaker (70 Amp or 80 Amp)
5	CB301, CB61	Circuit Breaker (80 Amp)
6	F408	Fuse (5 Amp)
7	CB304, CB305, CB67, CB68	Circuit Breaker (40 Amp)
8	S314	Radio Master Switch
9	K310, K315	Radio Master Contactor
10	A334, A335	Voltage Regulator
11		Day Night Switch
12	J420	External Power Receptacle
13		Alternator (1)
14	K403	External Power Contactor
15		(2)
16		(2)
17		Battery
18	K401	Battery Contactor
19	S315	Battery Switch
20		(2)
21	S316, S323	Alternator Switch
22	R304, R305	Shunt
23		(2)
24	CB395, CB393, CB52	Avionics Bus Tie Circuit Breaker (25 Amp)
25	CB60	Circuit Breaker (25 Amp)

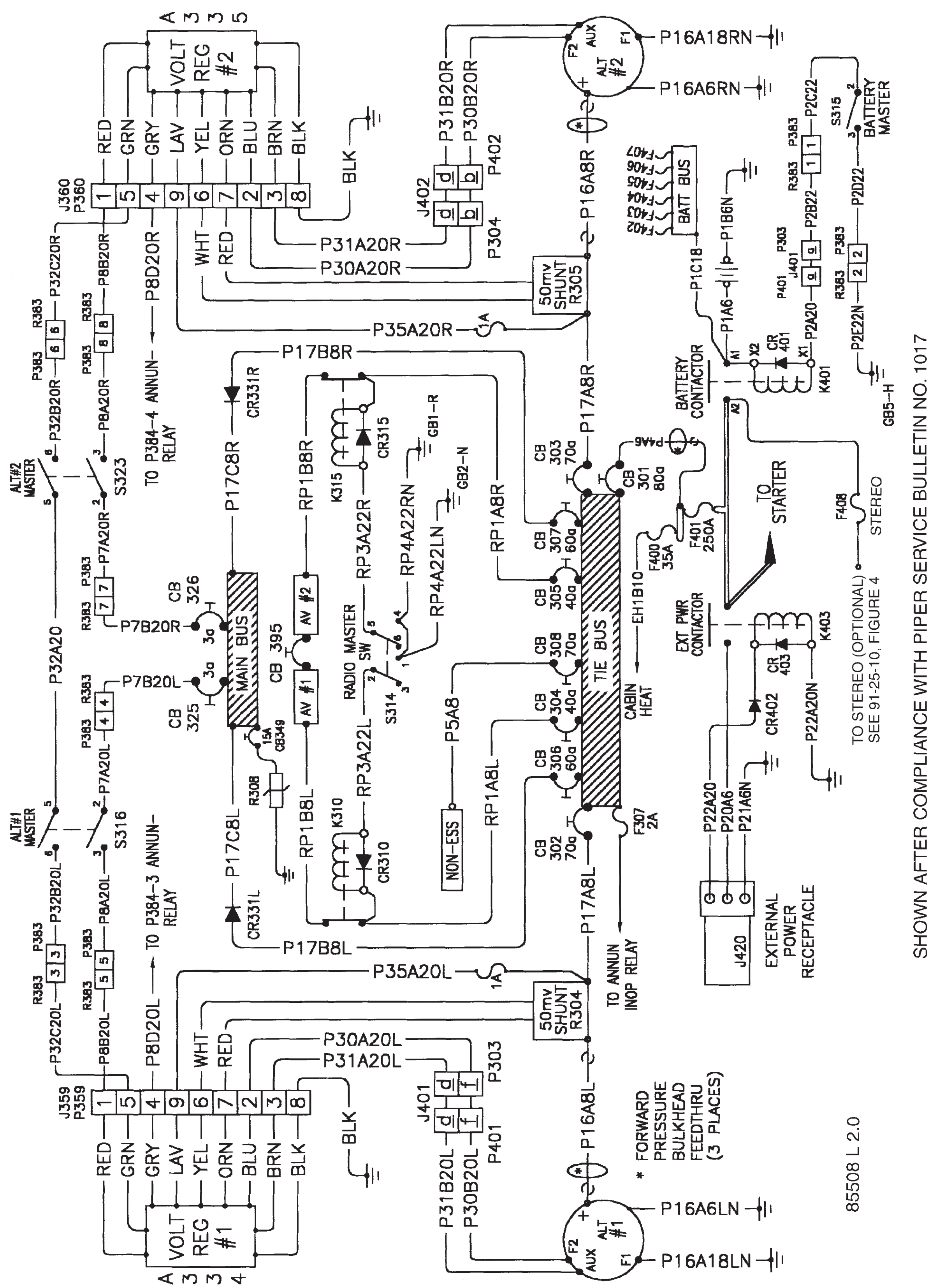
(1) Alternators supplied by Lycoming
(2) See Power Monitor and Meters



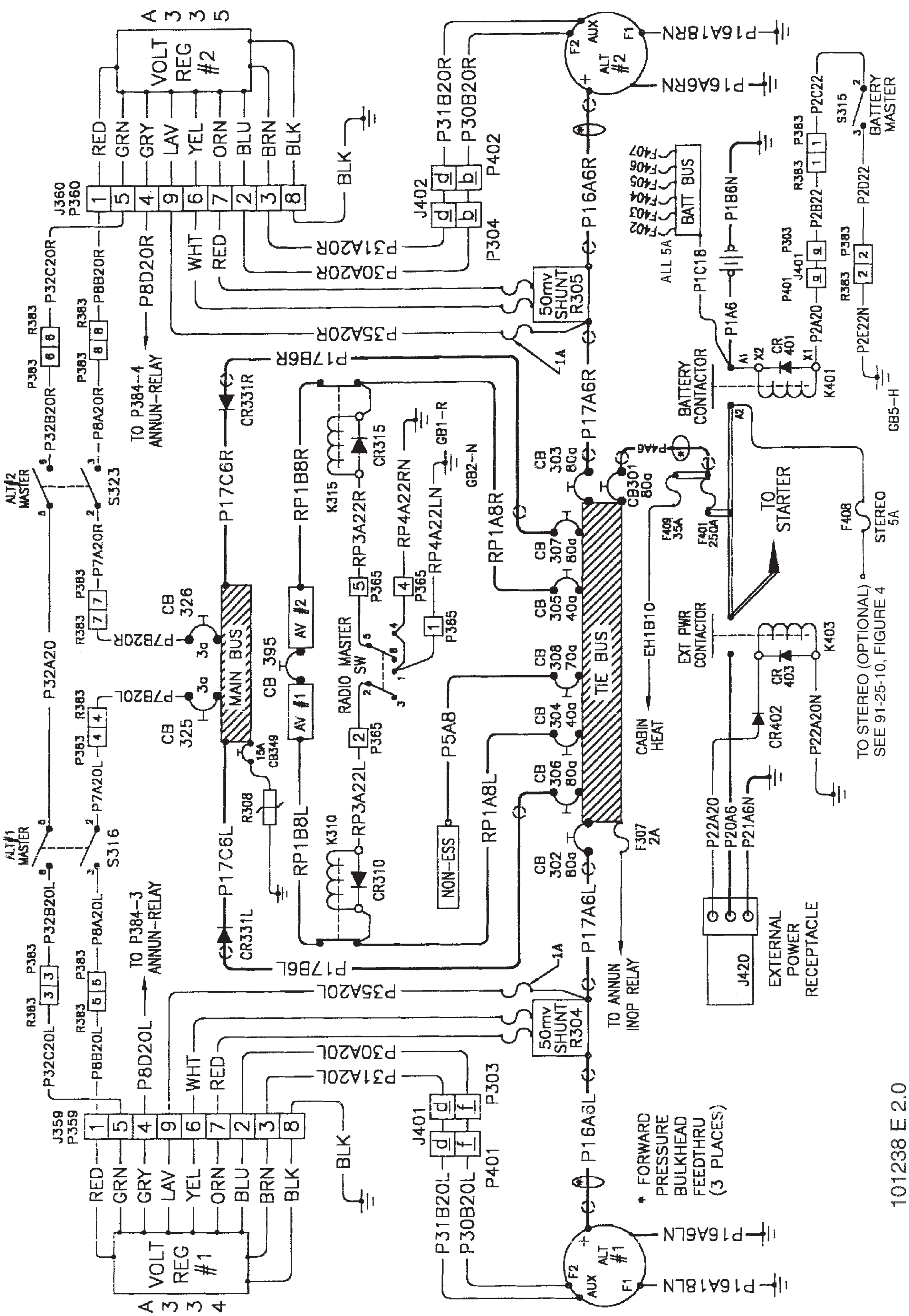
Power Distribution and Control
Figure 1 (Sheet 1 of 14)



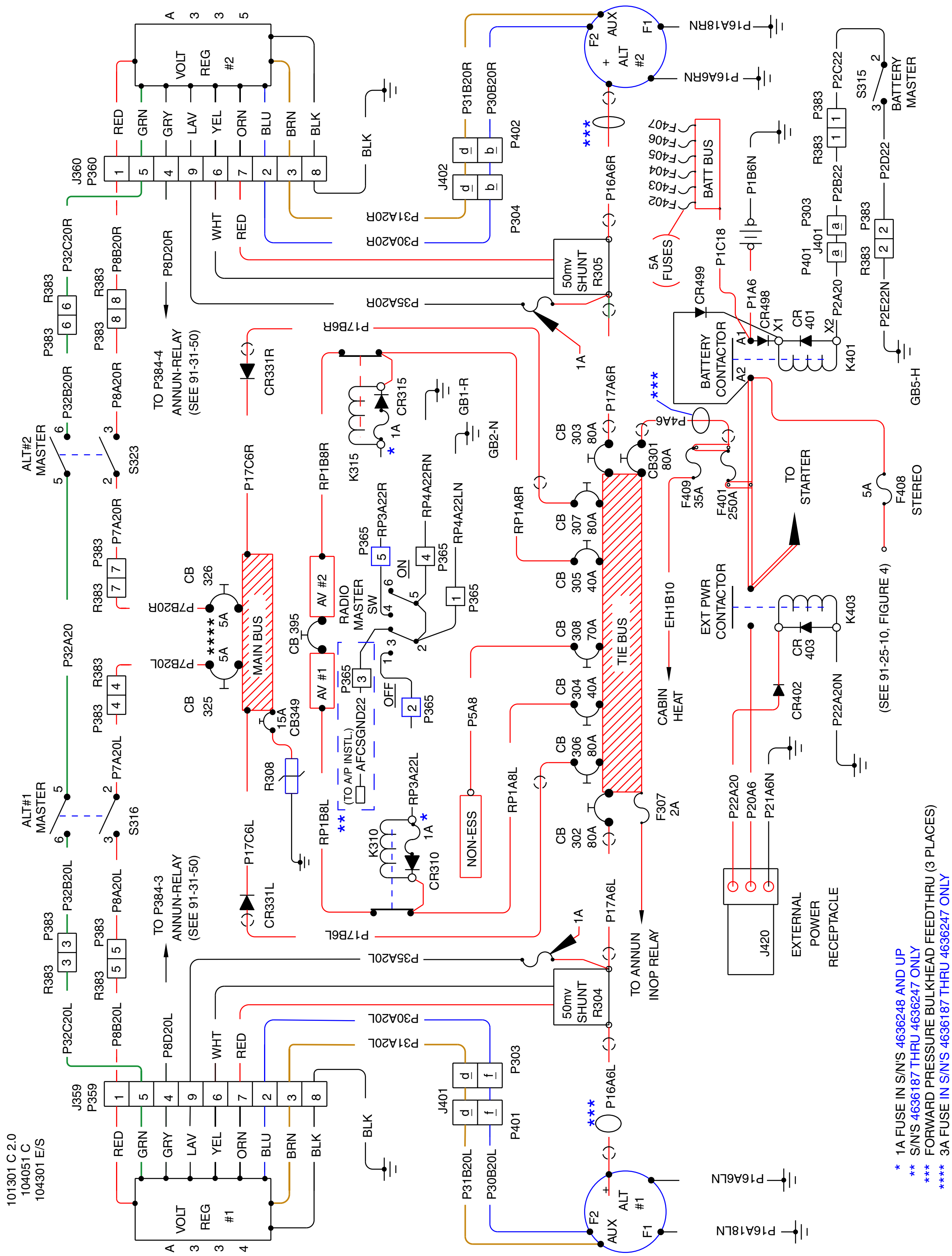
Power Distribution and Control
Figure 1 (Sheet 2 of 14)



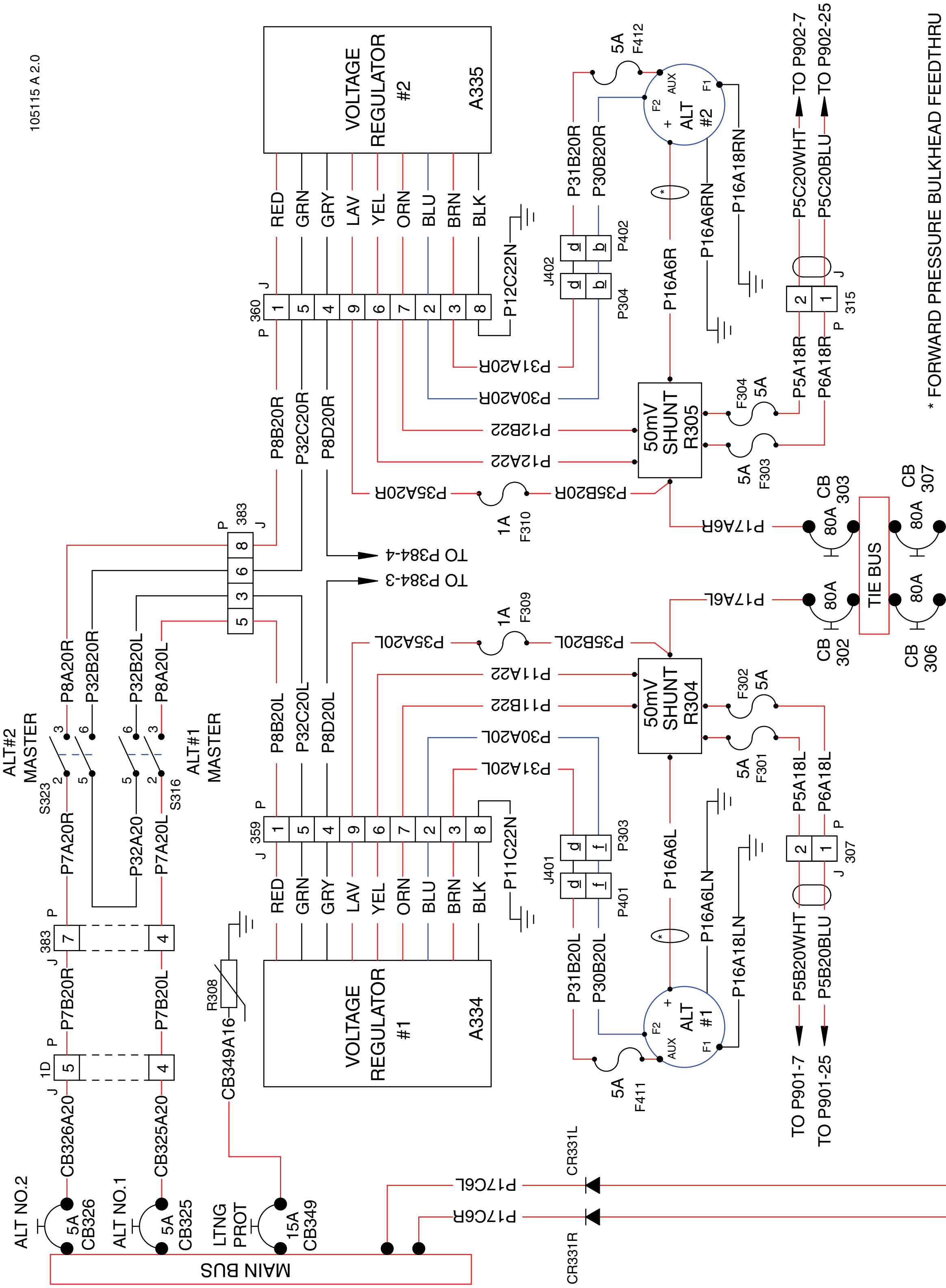
Power Distribution and Control
Figure 1 (Sheet 3 of 14)



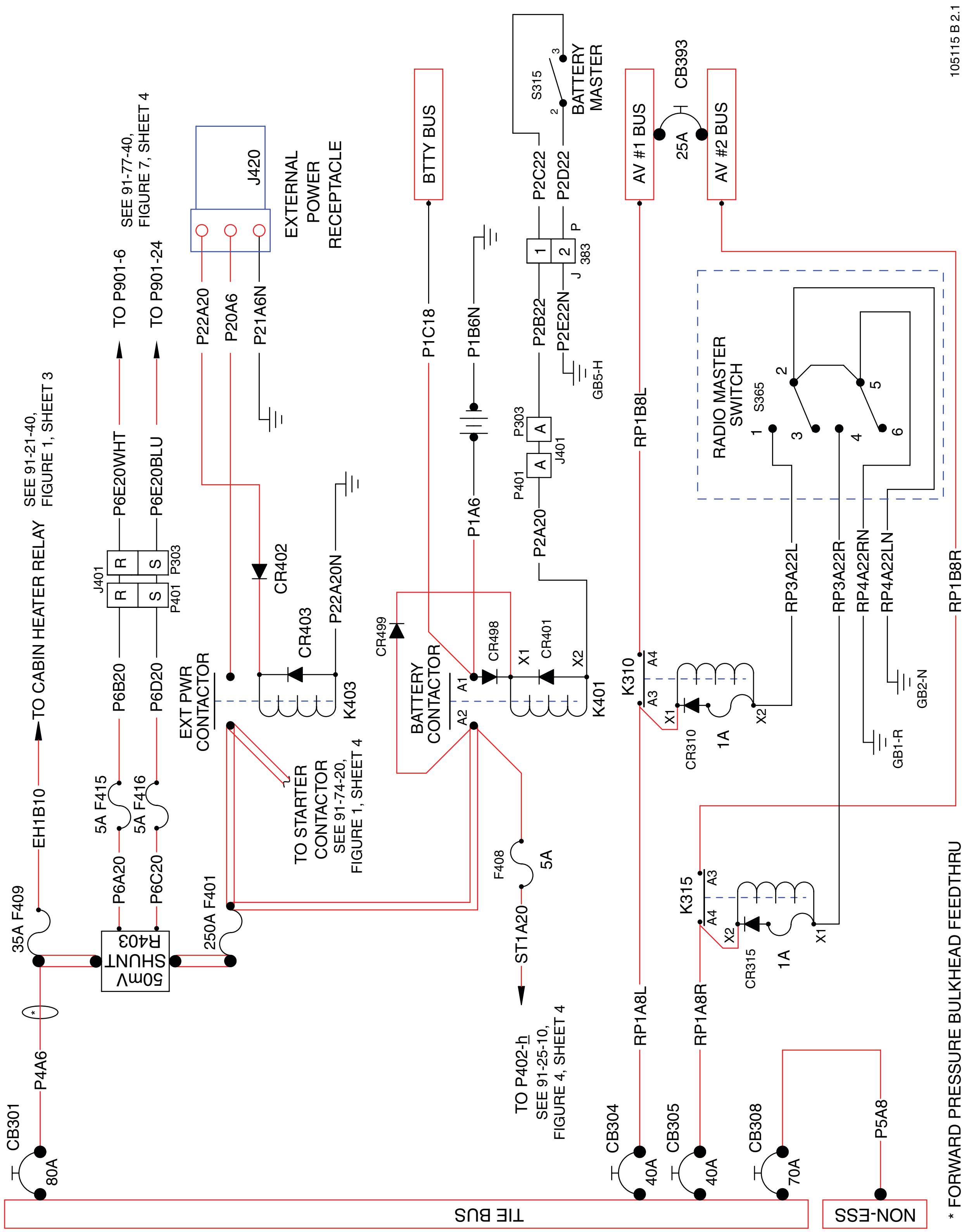
Power Distribution and Control
Figure 1 (Sheet 4 of 14)



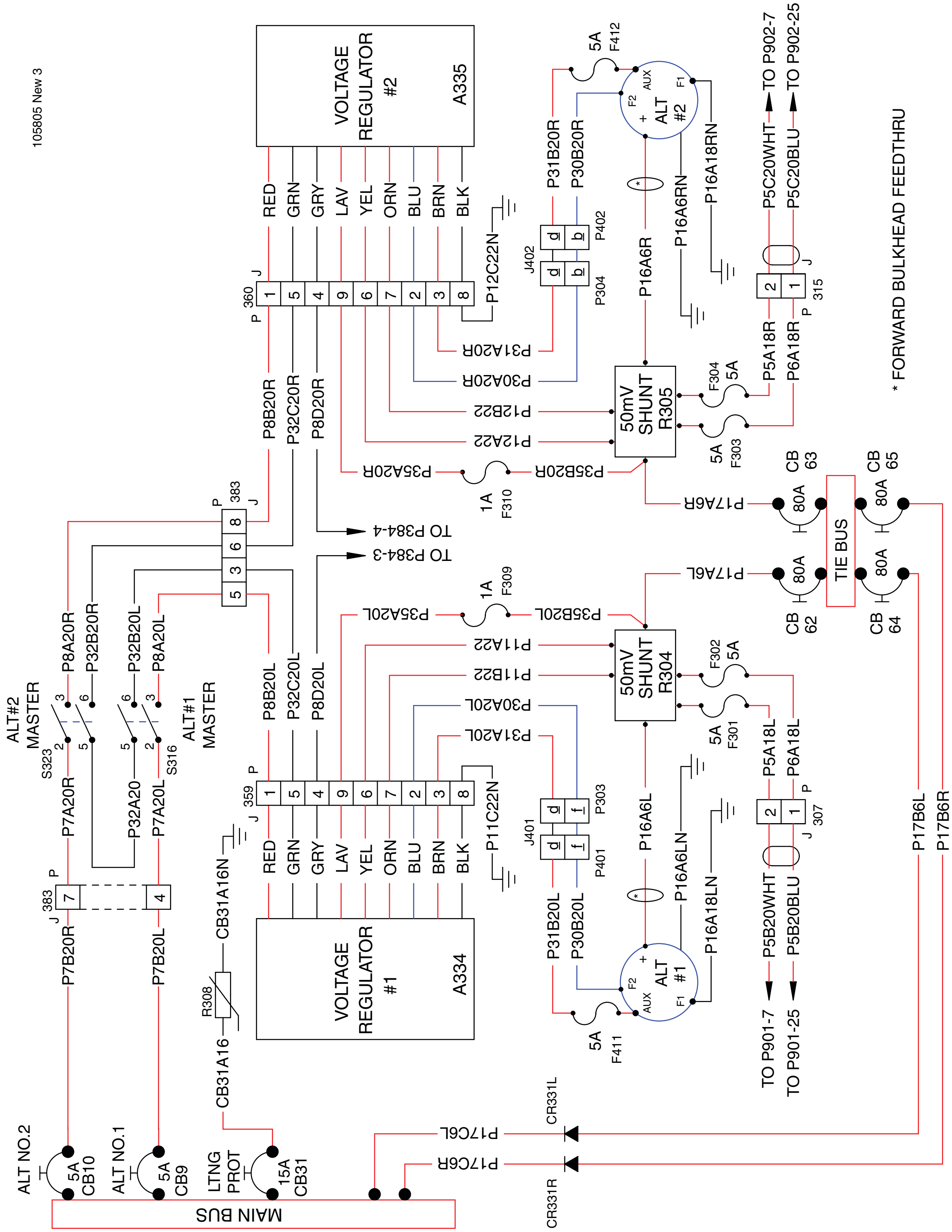
Power Distribution and Control
Figure 1 (Sheet 5 of 14)



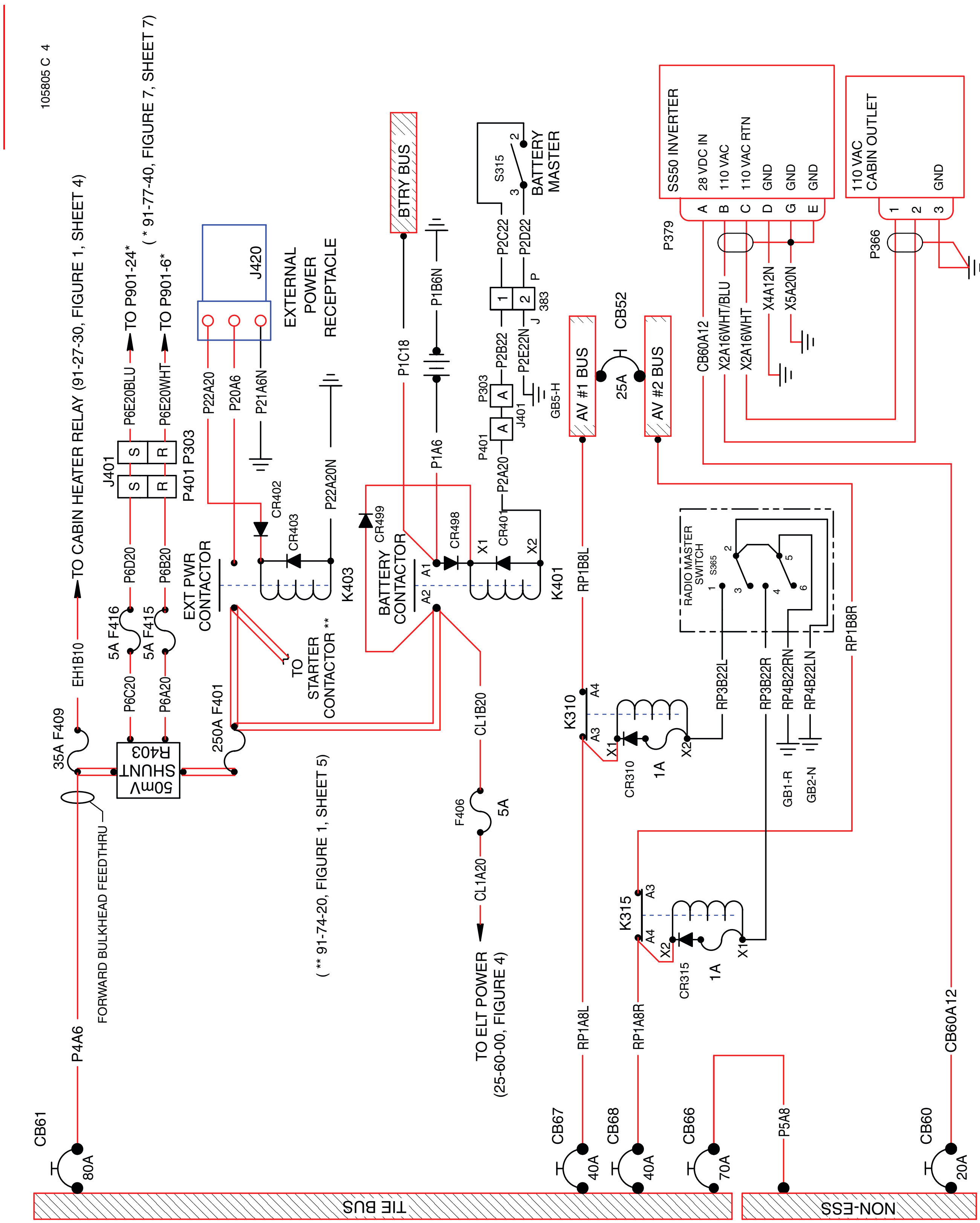
Power Distribution and Control
Figure 1 (Sheet 6 of 14)



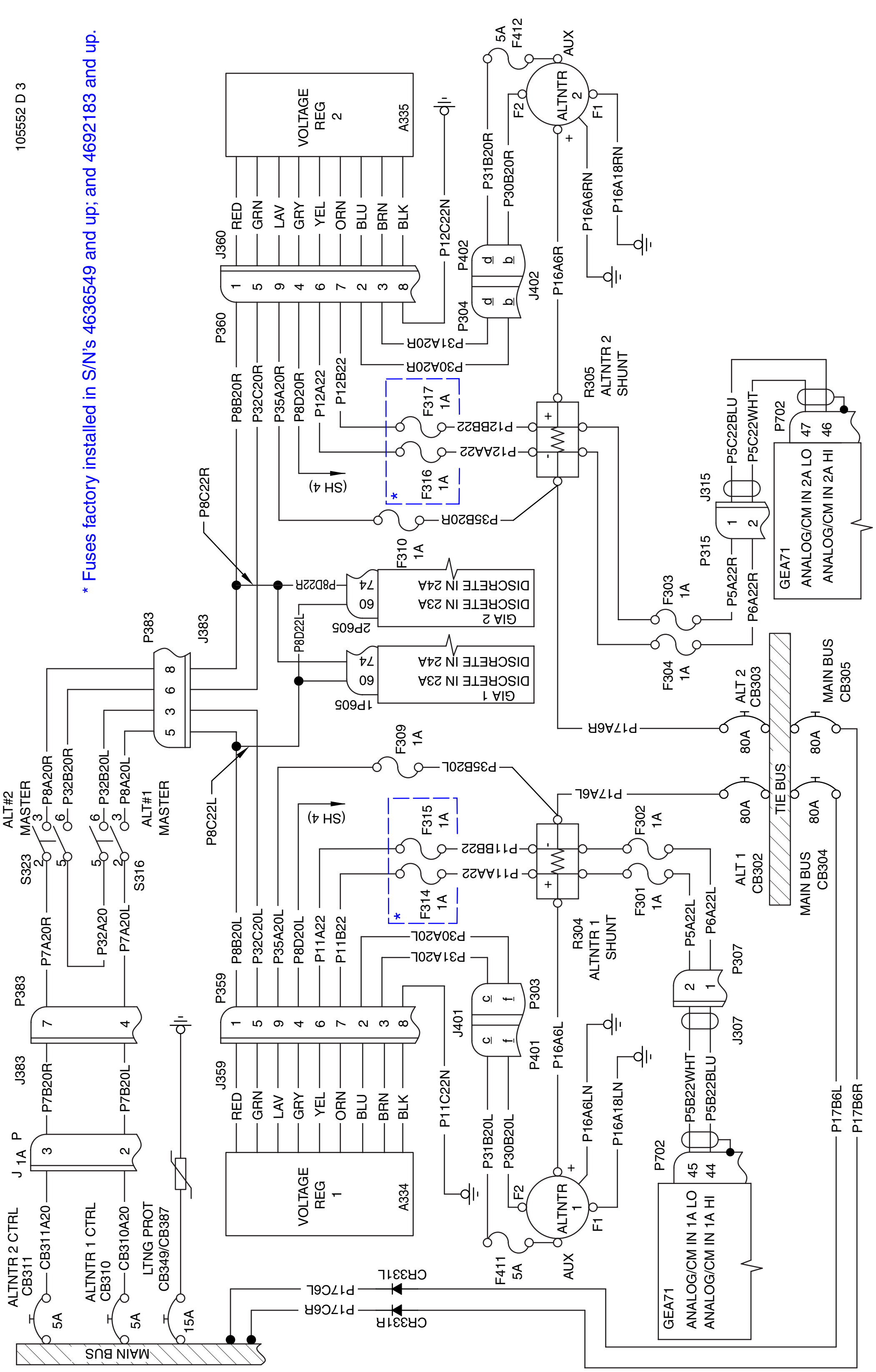
Power Distribution and Control
Figure 1 (Sheet 7 of 14)



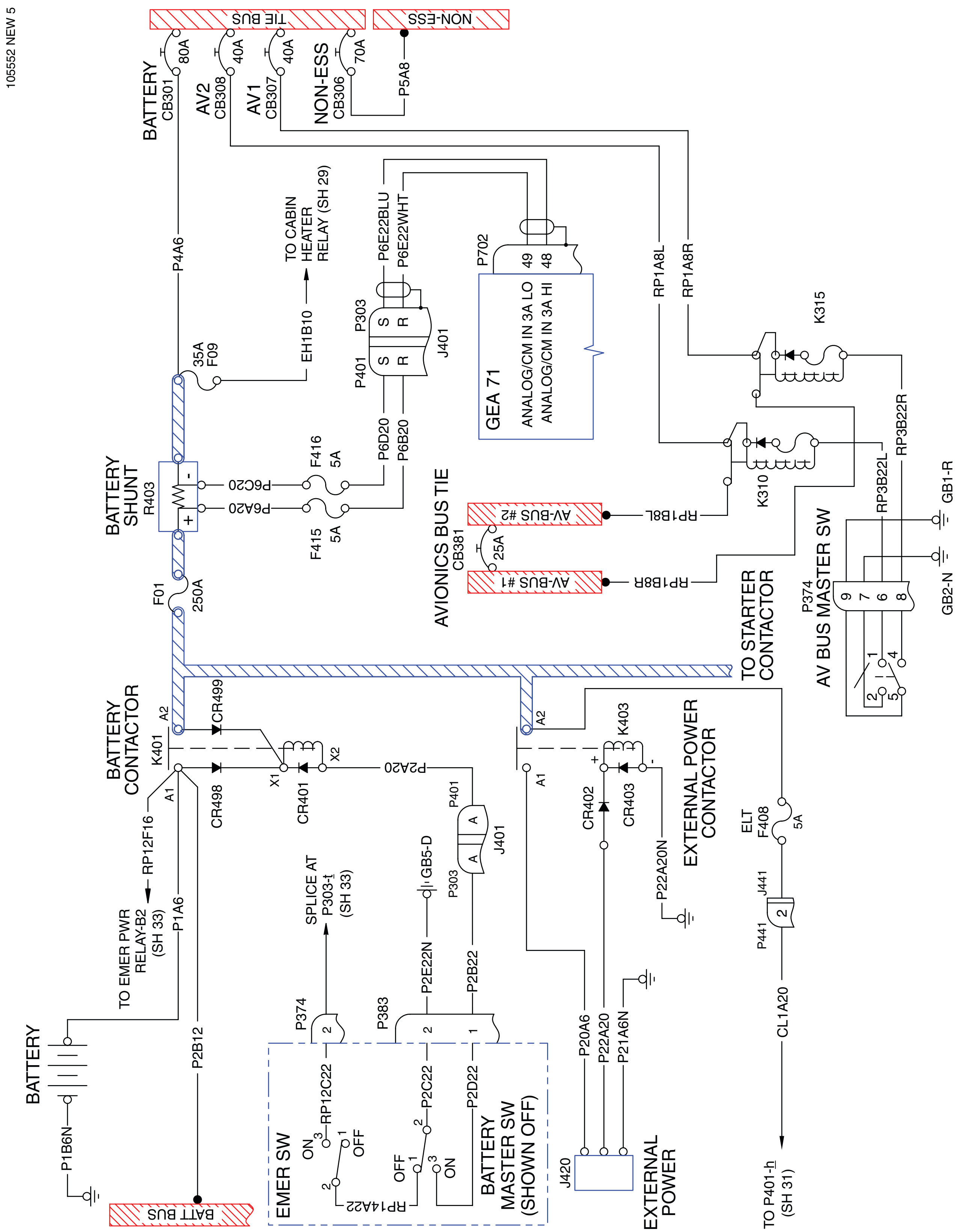
Power Distribution and Control
Figure 1 (Sheet 8 of 14)



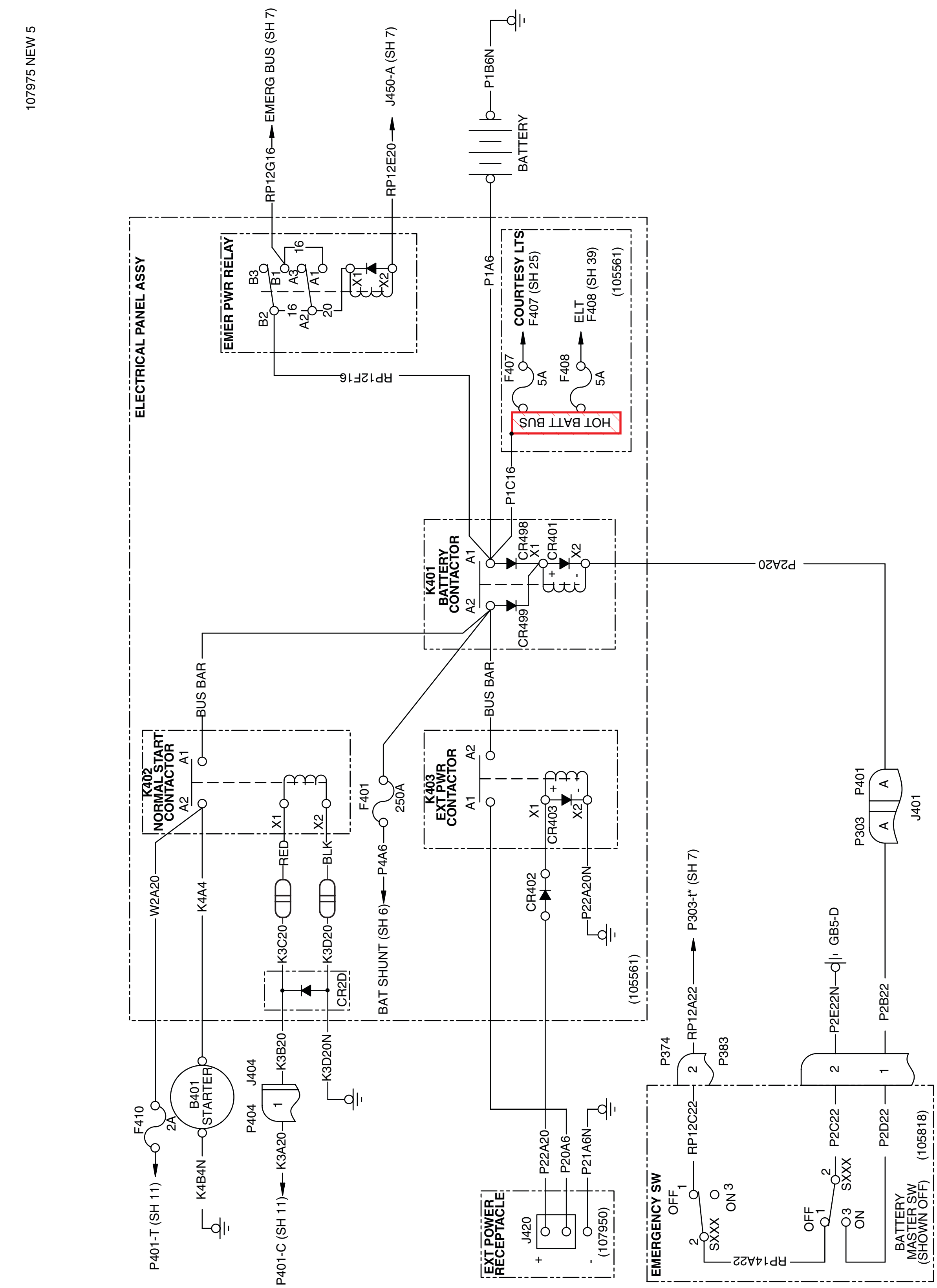
Power Distribution and Control
Figure 1 (Sheet 9 of 14)



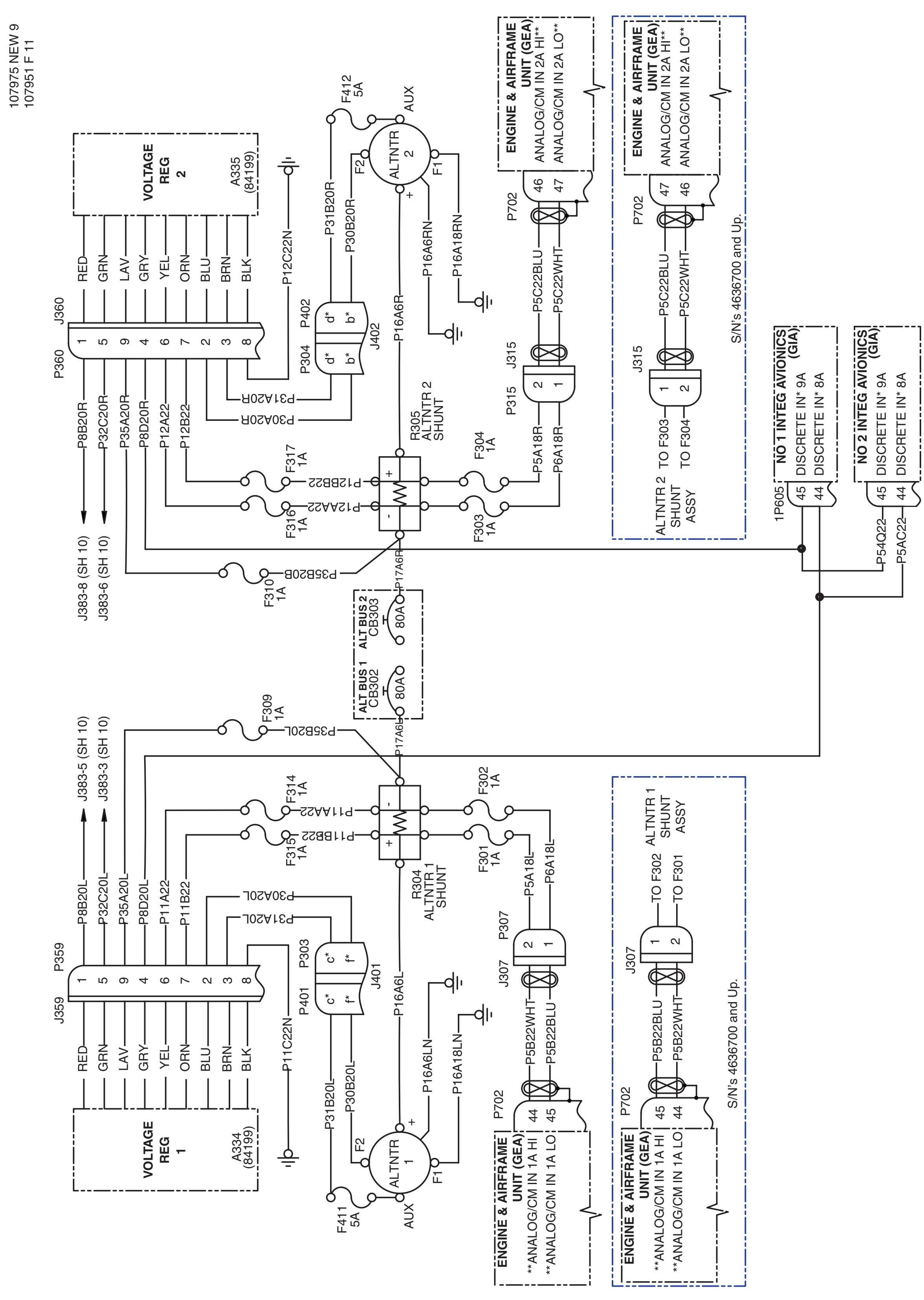
Power Distribution and Control
Figure 1 (Sheet 10 of 14)



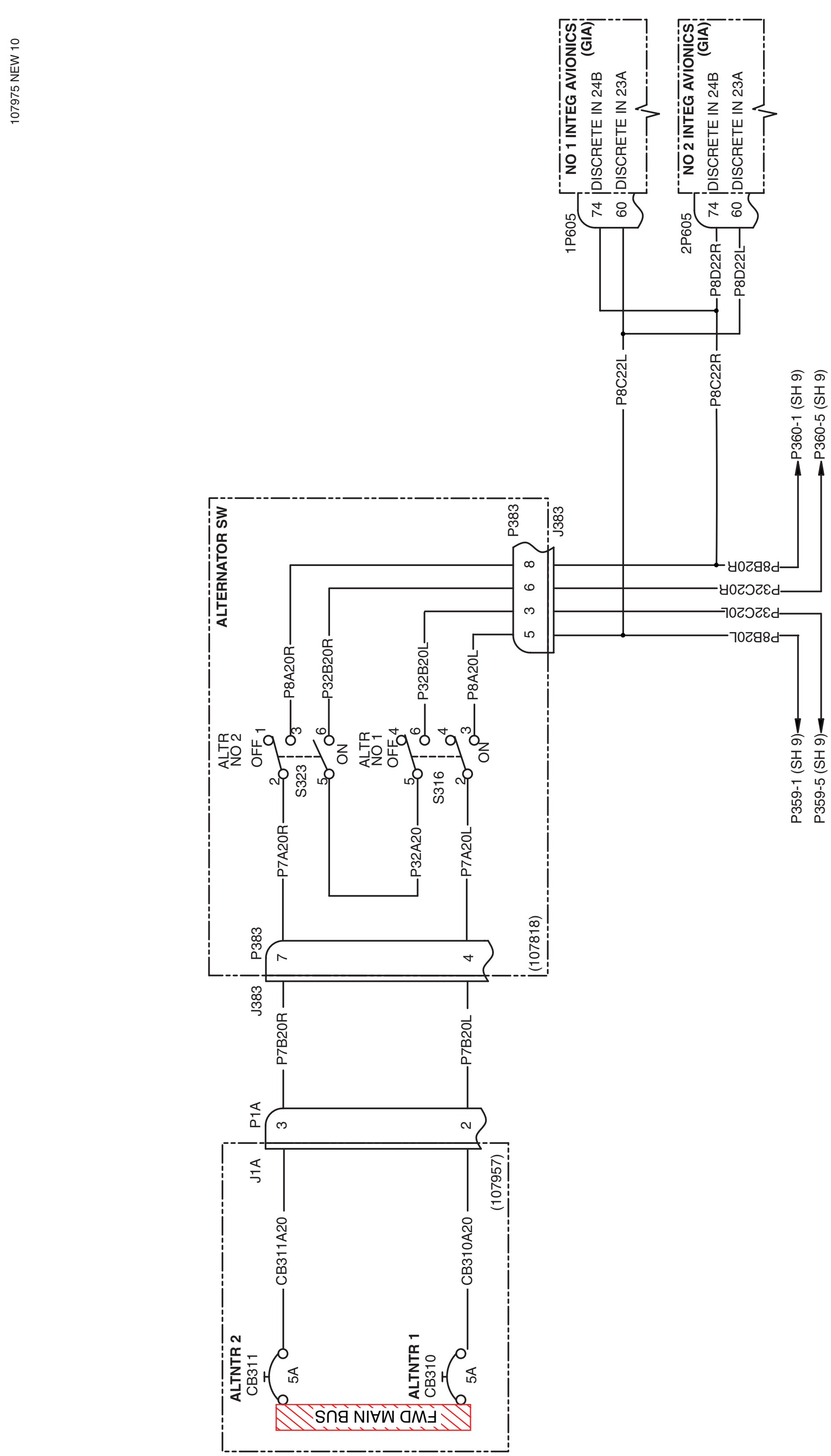
Power Distribution and Control
Figure 1 (Sheet 11 of 14)



Power Distribution and Control
Figure 1 (Sheet 12 of 14)



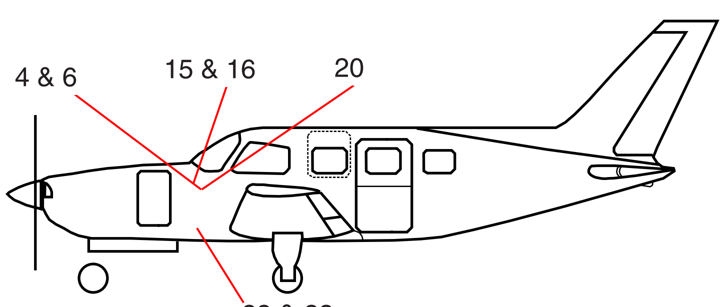
Power Distribution and Control
Figure 1 (Sheet 13 of 14)



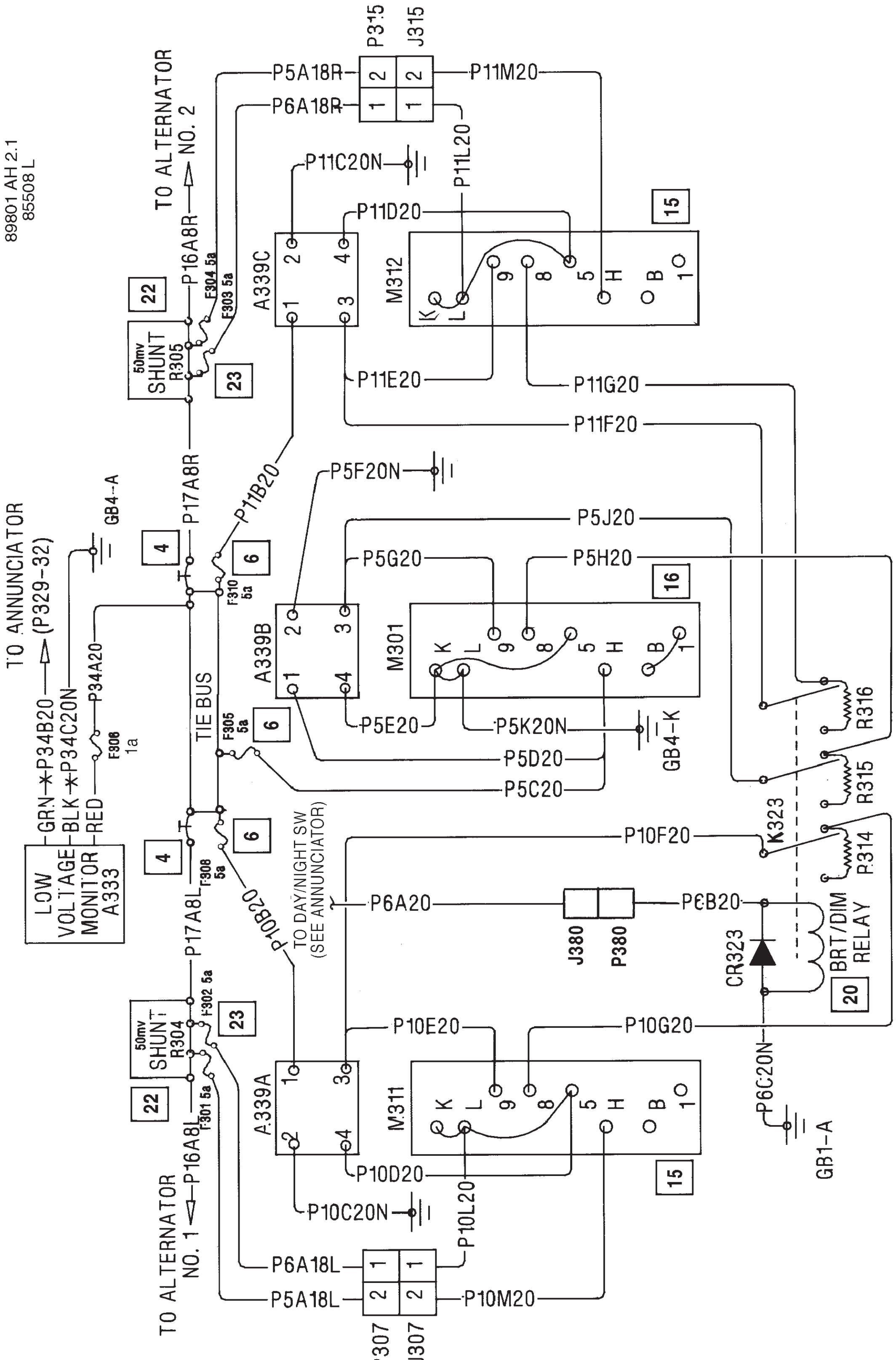
Power Distribution and Control
Figure 1 (Sheet 14 of 14)

Item #	Designation	Part Number	Description
1 - 3*			
4	CB302, CB303		Circuit Breaker (70 Amp or 80 Amp)
5*			
6	F408		Fuse (5 Amp)
7 - 14*			
15	M311, M312		Ammeters
16	M301		Volt Meter
17 - 19*			
20	K323		Day/Night (Bright Dim) Relay
21*			
22	R304, R305		Shunt
23	F301 - F304		Fuse (5 Amp)

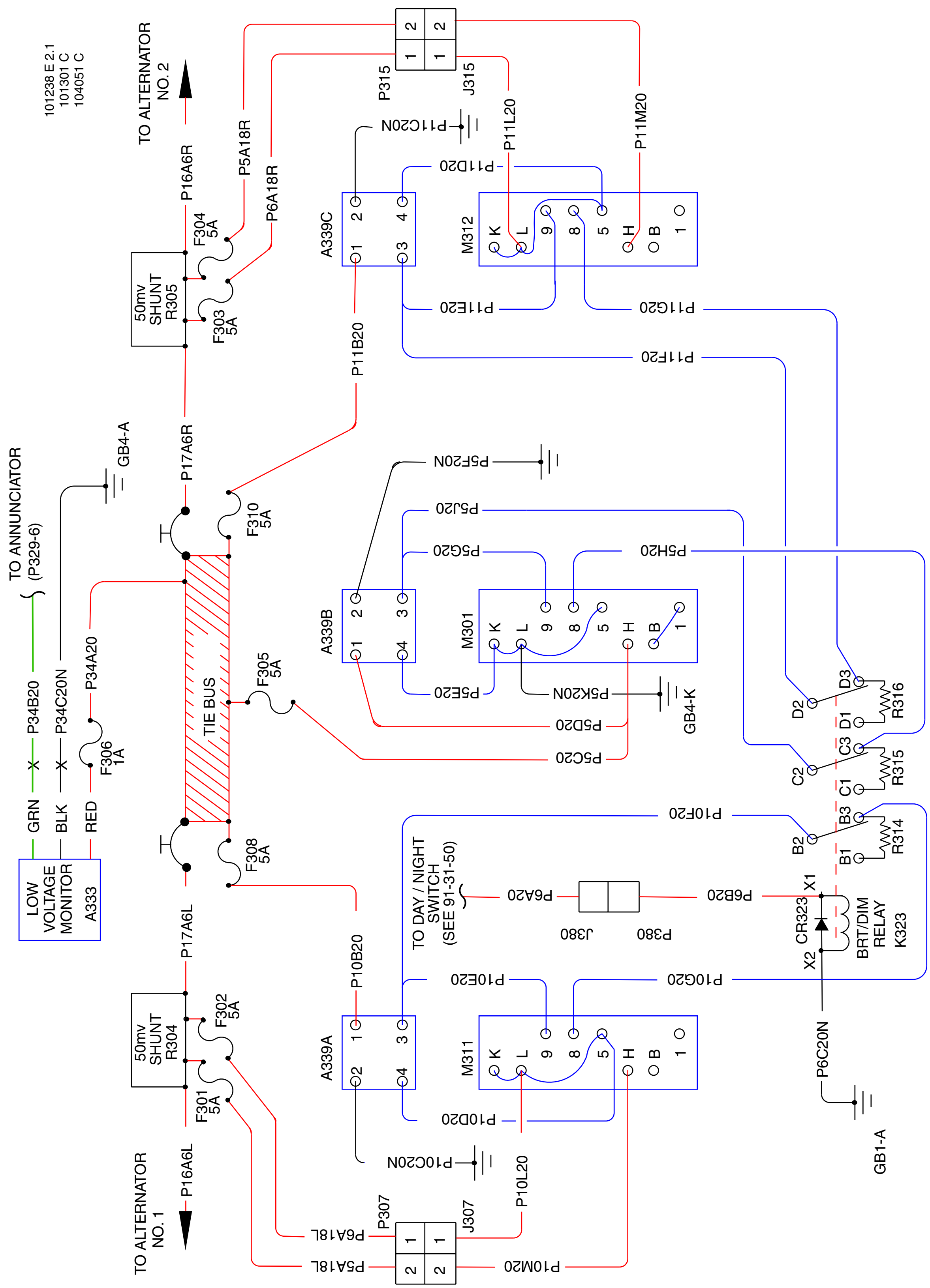
* See Power Distribution and Control



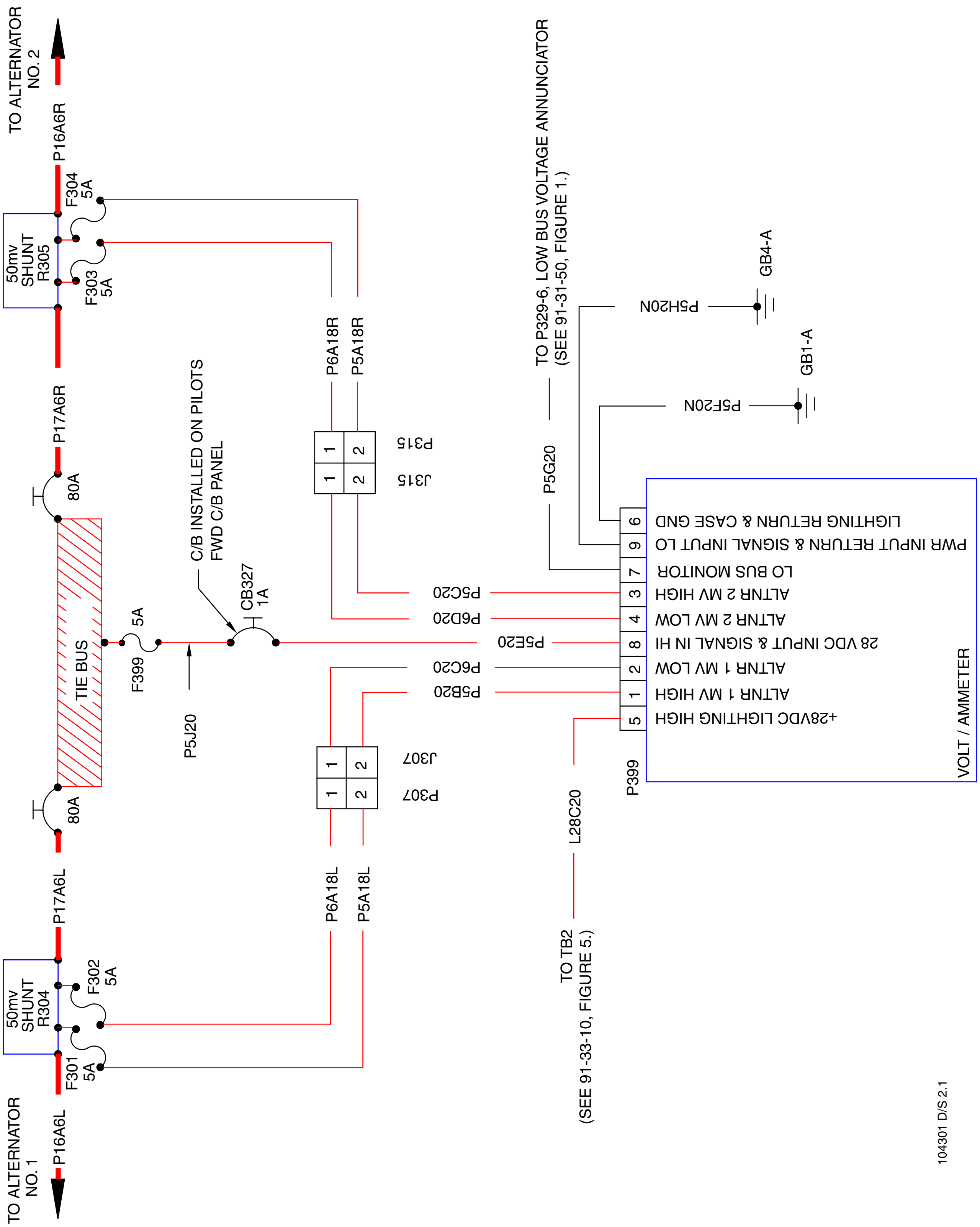
Power Monitor and Meters
Figure 2 (Sheet 1 of 5)



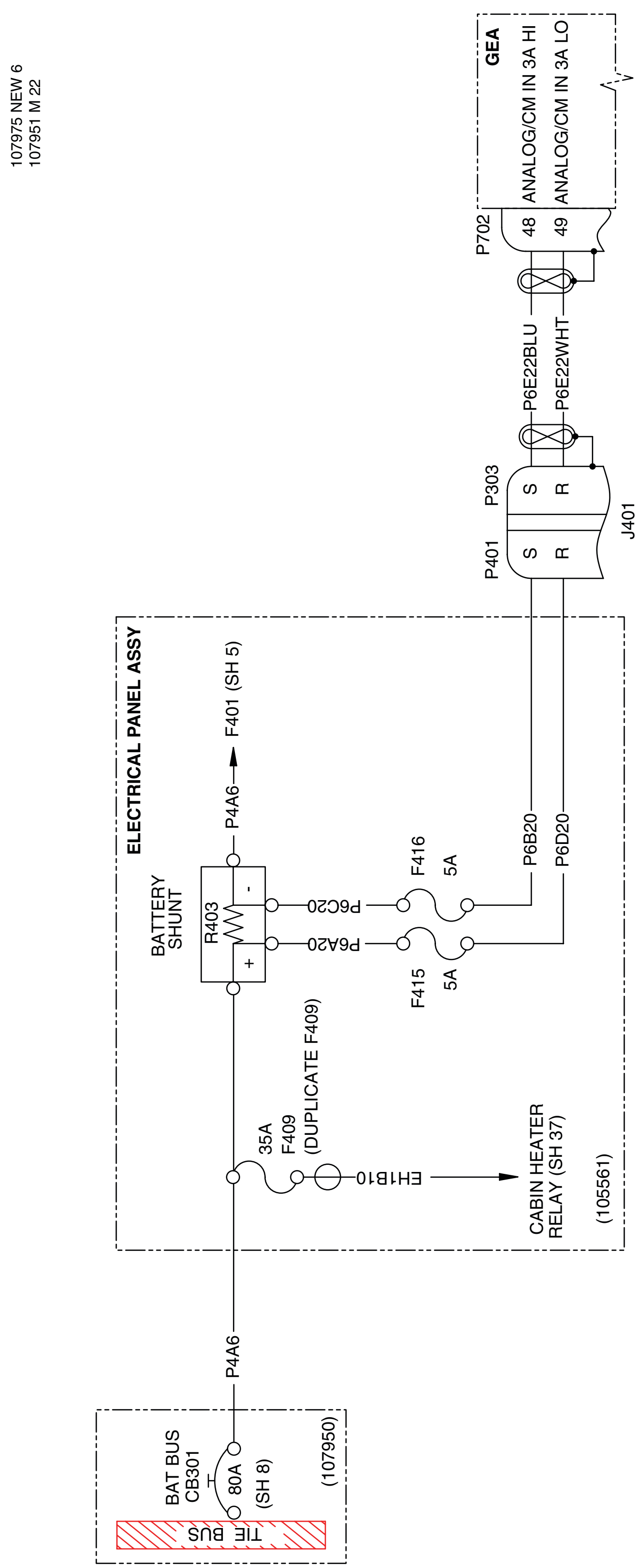
Power Monitor and Meters
Figure 2 (Sheet 2 of 5)



Power Monitor and Meters
Figure 2 (Sheet 3 of 5)



Power Monitor and Meters
Figure 2 (Sheet 4 of 5)



NOTE: SEE 105552 SH 5 for S/N's 4636460, 4636463–4636651 less 4636481 and 4636633; and 4692134 and up, less 4692141, 4692149, 4692153

Power Monitor and Meters
Figure 2 (Sheet 5 of 5)

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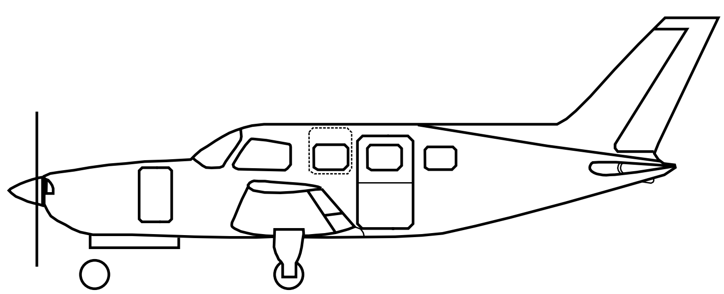
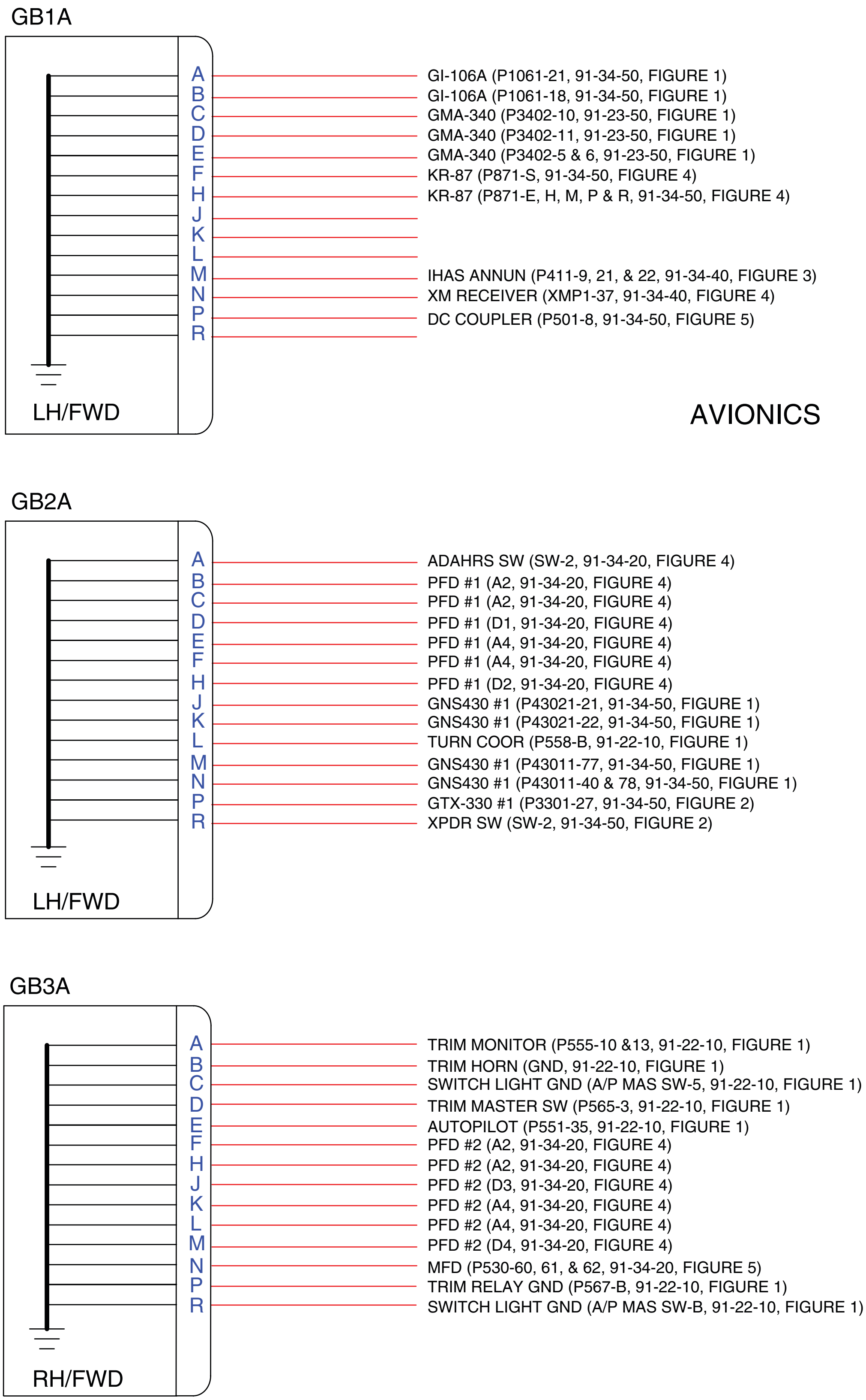
Item #	Designation	Description

Information Pending - See Parts Catalog

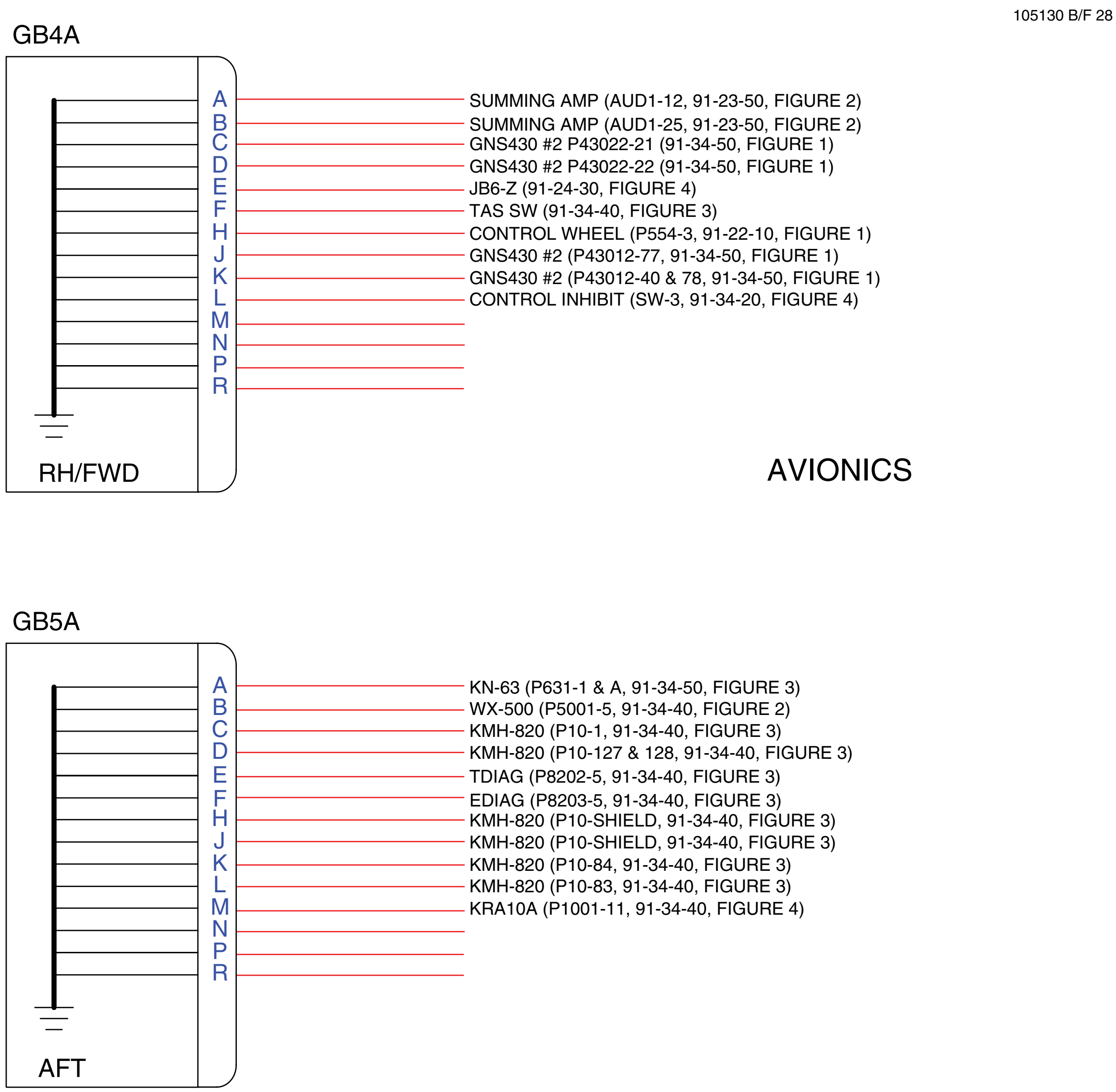


Ground Blocks
Figure 3 (Sheet 1 of 10)

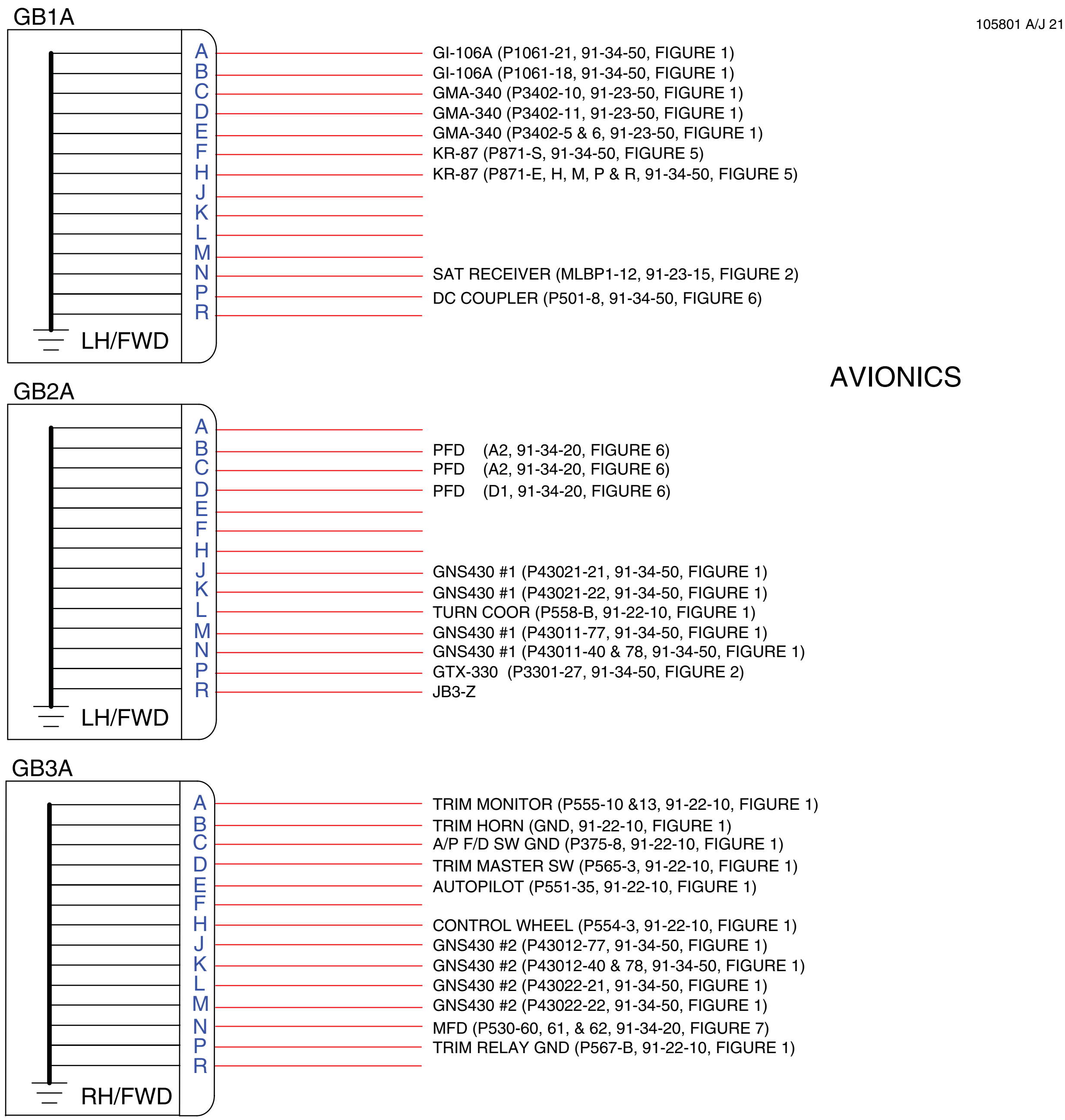
105130 B/F 28



Ground Blocks
Figure 3 (Sheet 2 of 10)

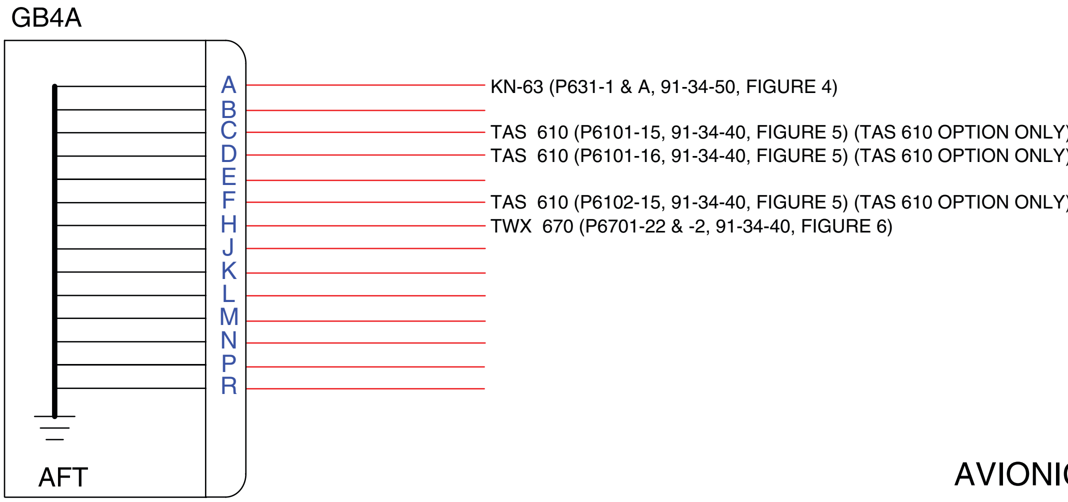


Ground Blocks
Figure 3 (Sheet 3 of 10)

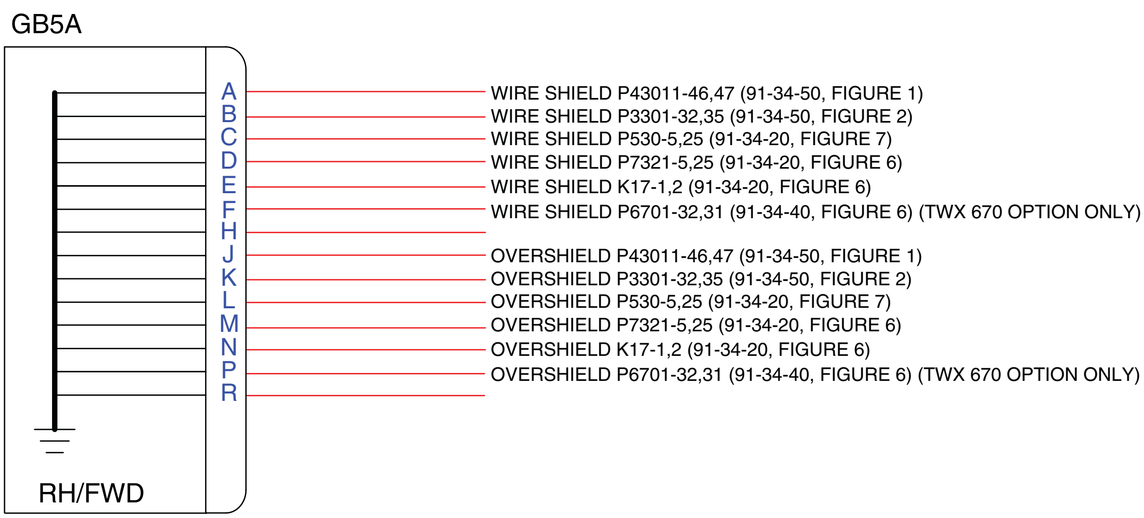


Ground Blocks
Figure 3 (Sheet 4 of 10)

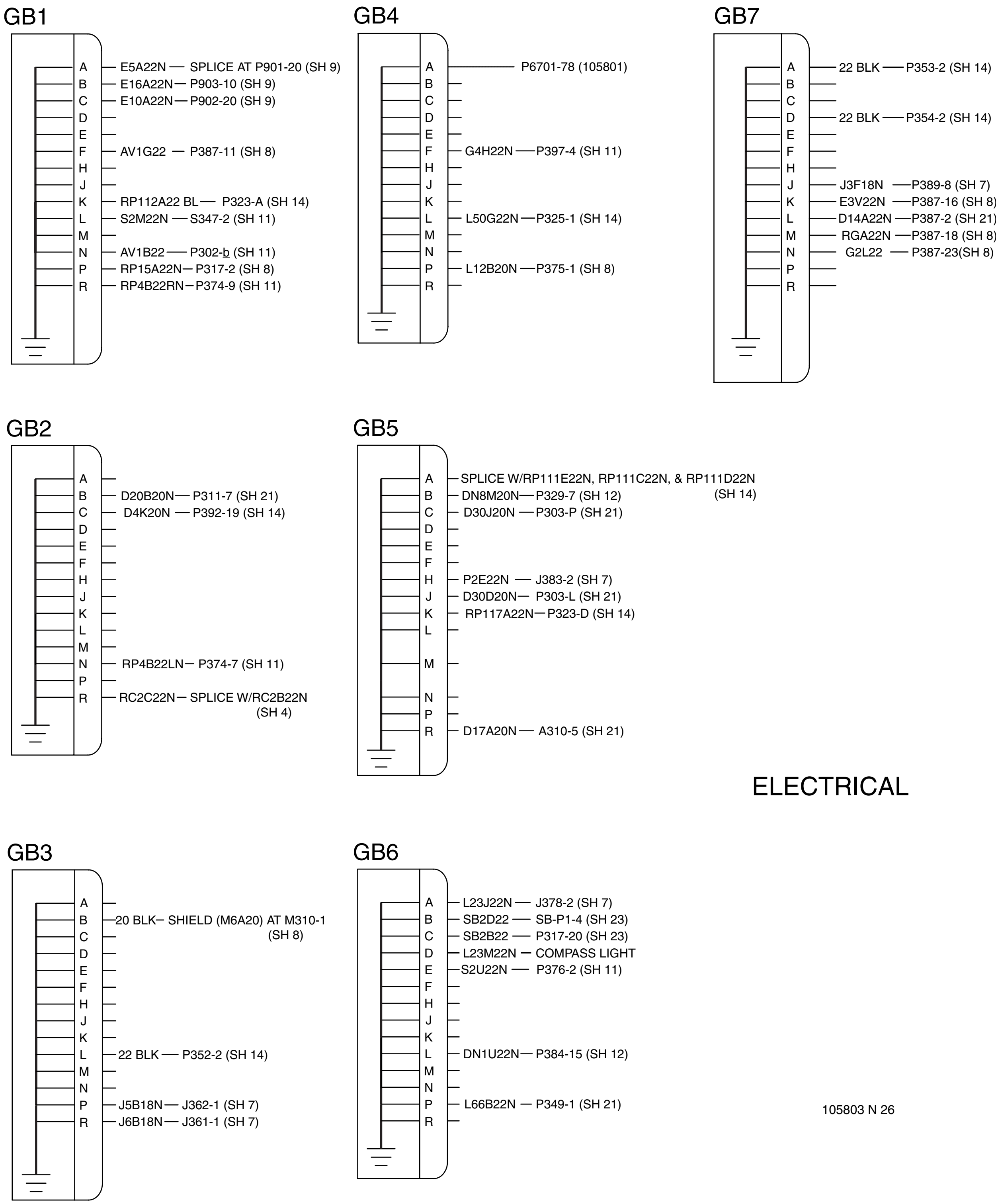
105801 A/J 21



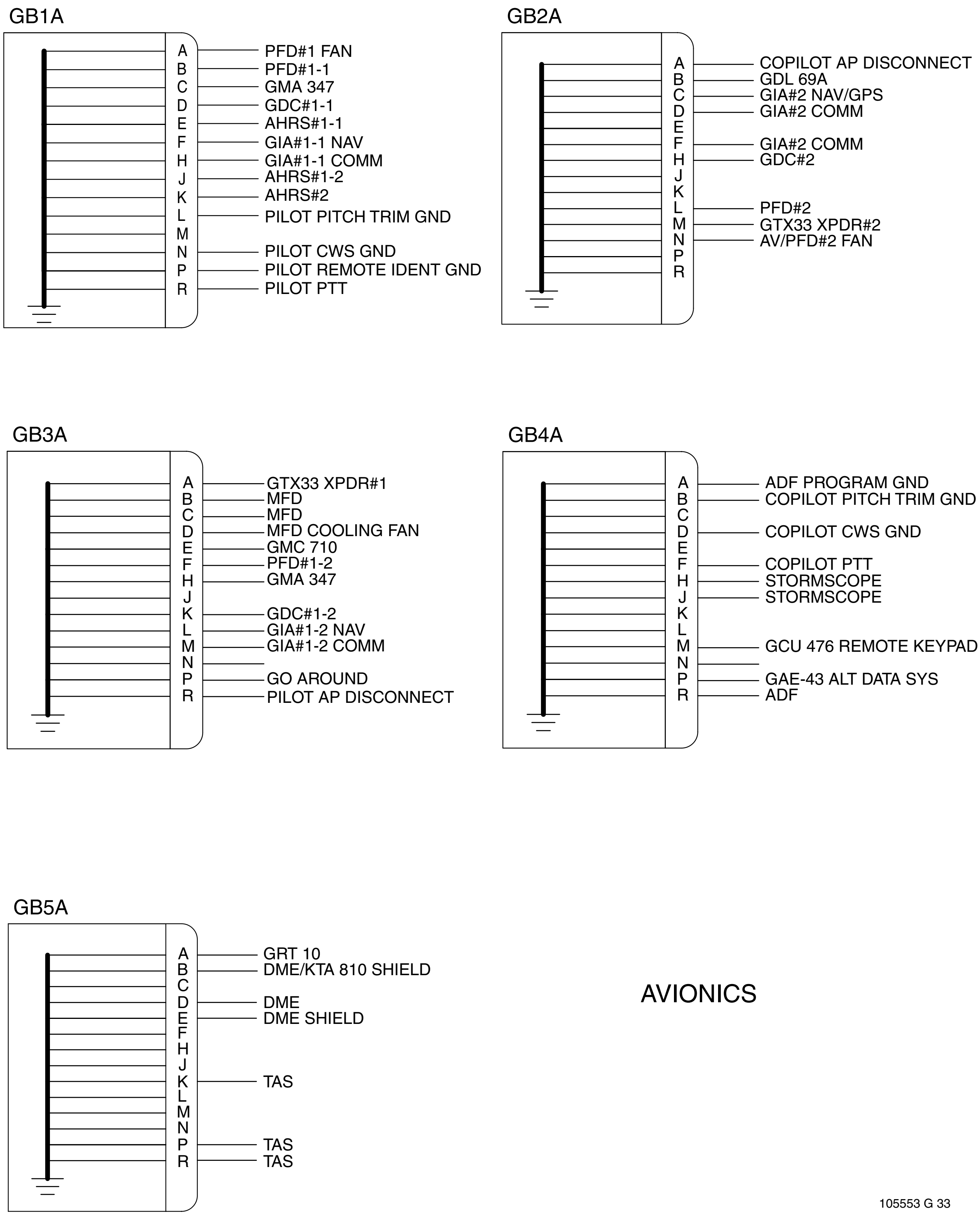
AVIONICS



Ground Blocks
Figure 3 (Sheet 5 of 10)



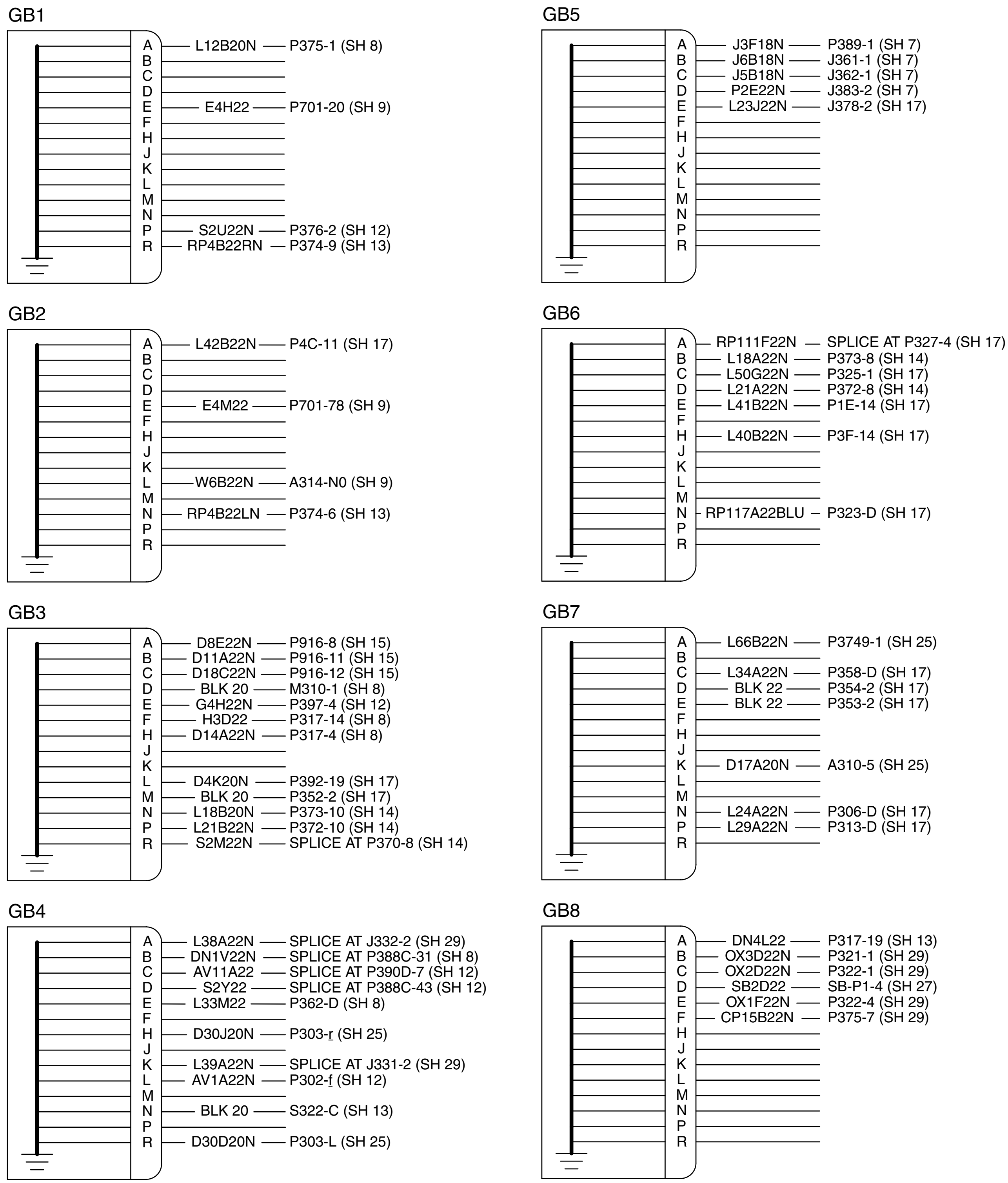
Ground Blocks
Figure 3 (Sheet 6 of 10)



Ground Blocks
Figure 3 (Sheet 7 of 10)

ELECTRICAL

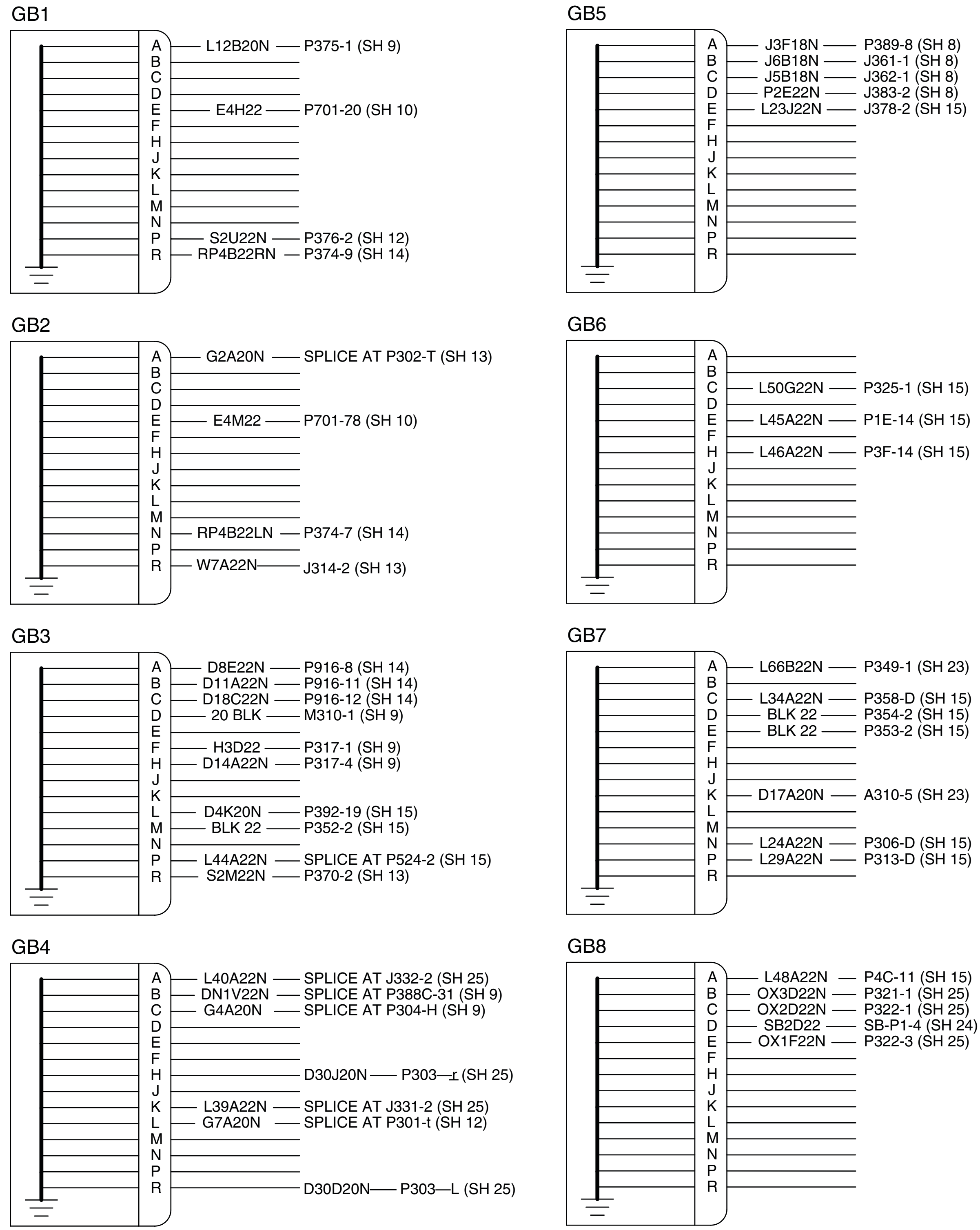
105551 U 30



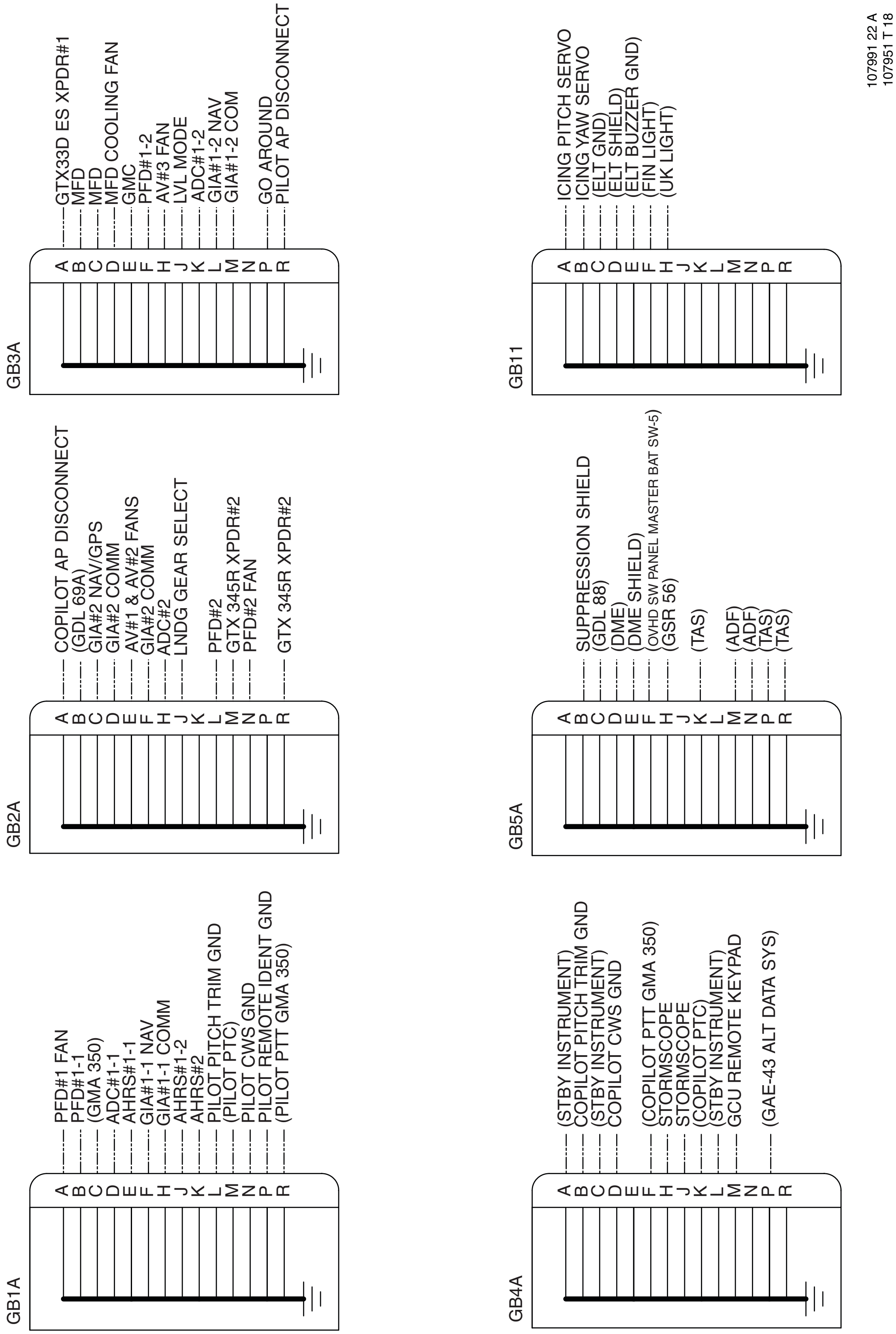
Ground Blocks
Figure 3 (Sheet 8 of 10)

ELECTRICAL

107951 B 26



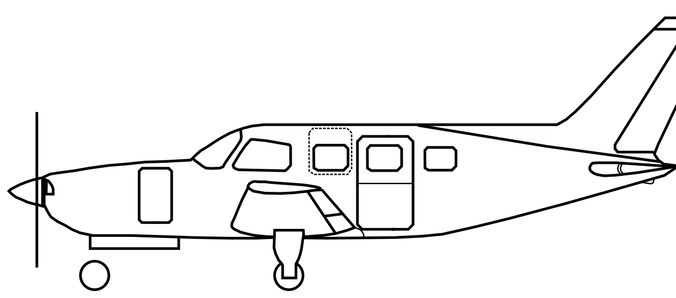
Ground Blocks
Figure 3 (Sheet 9 of 10)



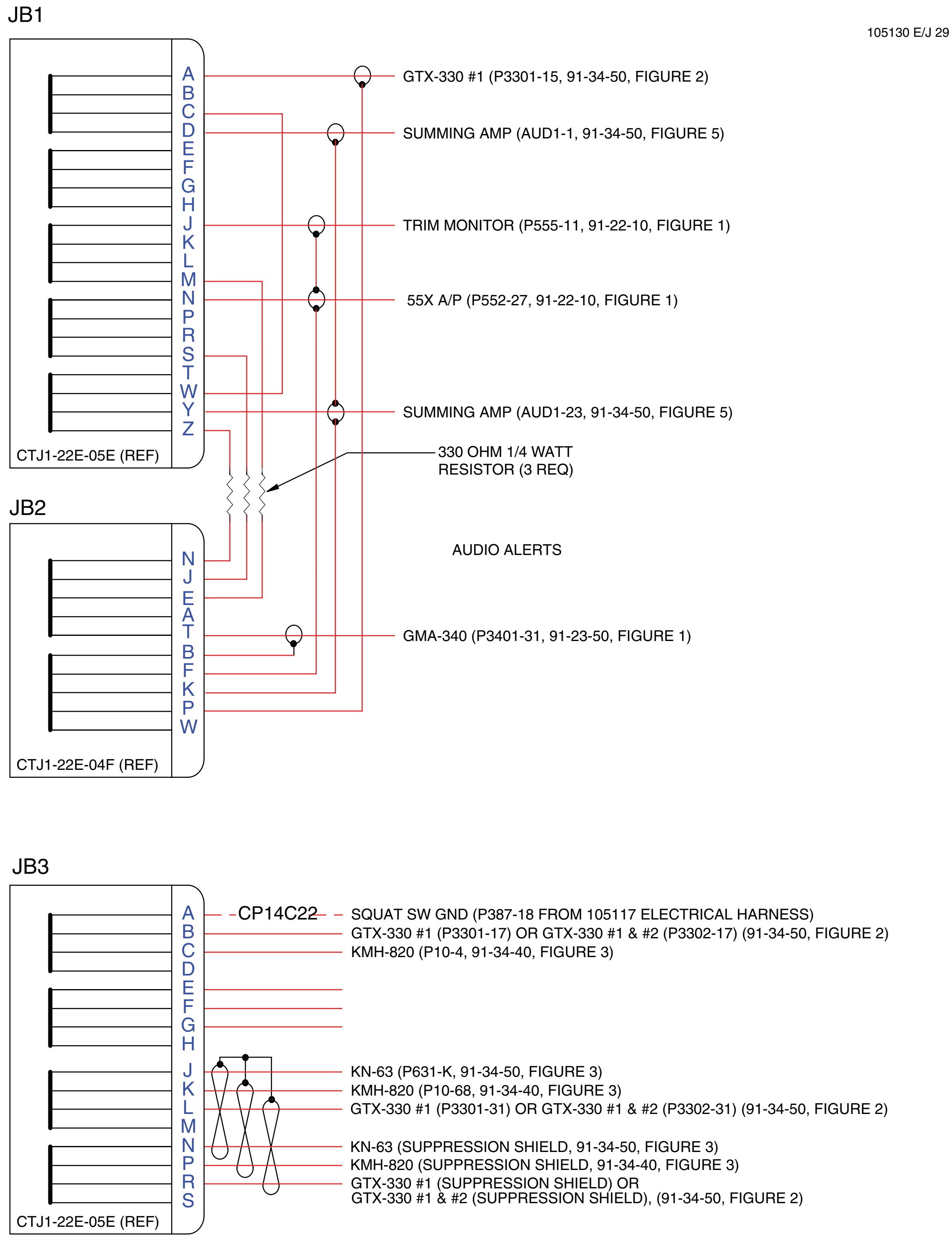
Ground Blocks
Figure 3 (Sheet 10 of 10)

Item #	Designation	Description

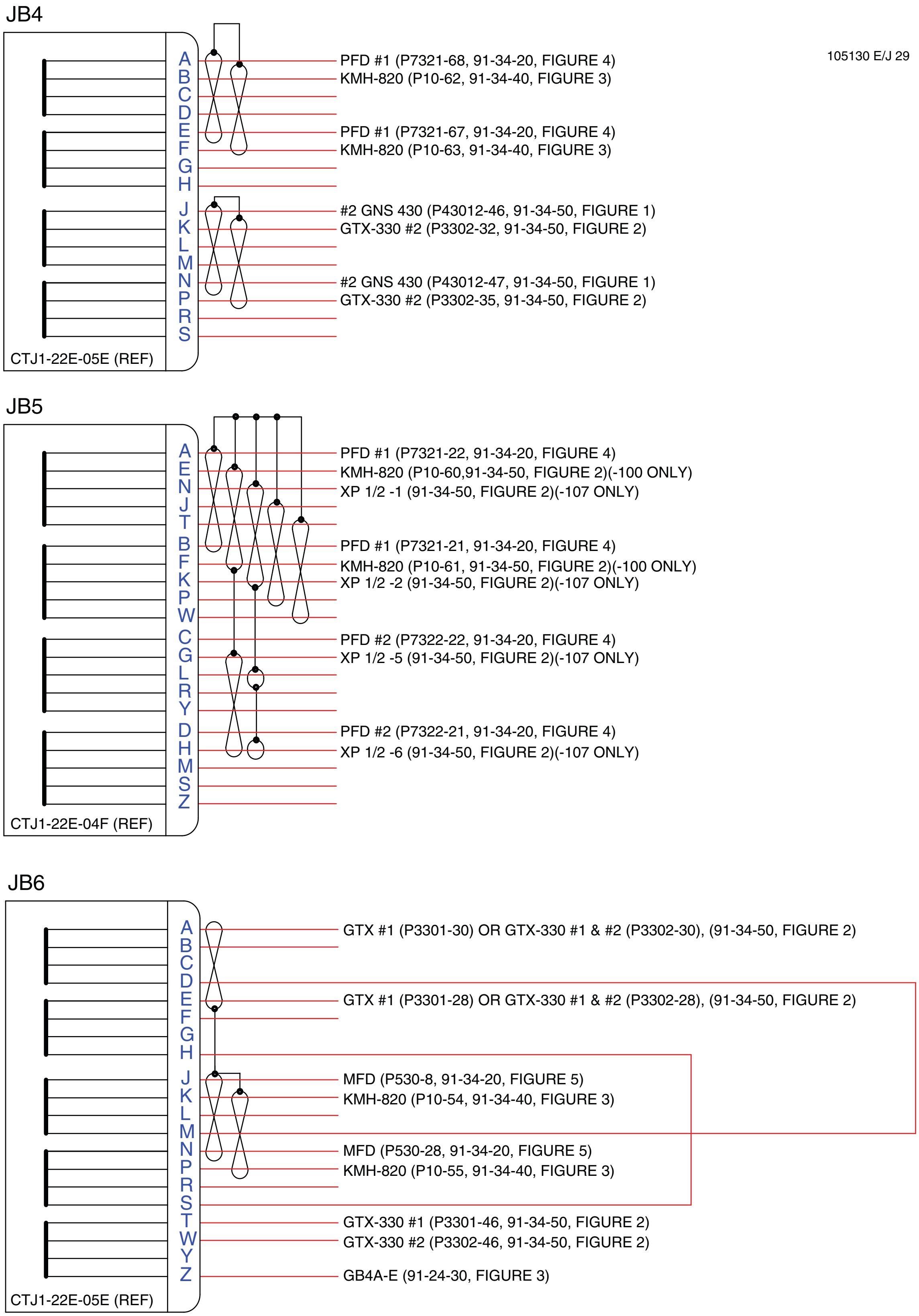
Information Pending - See Parts Catalog



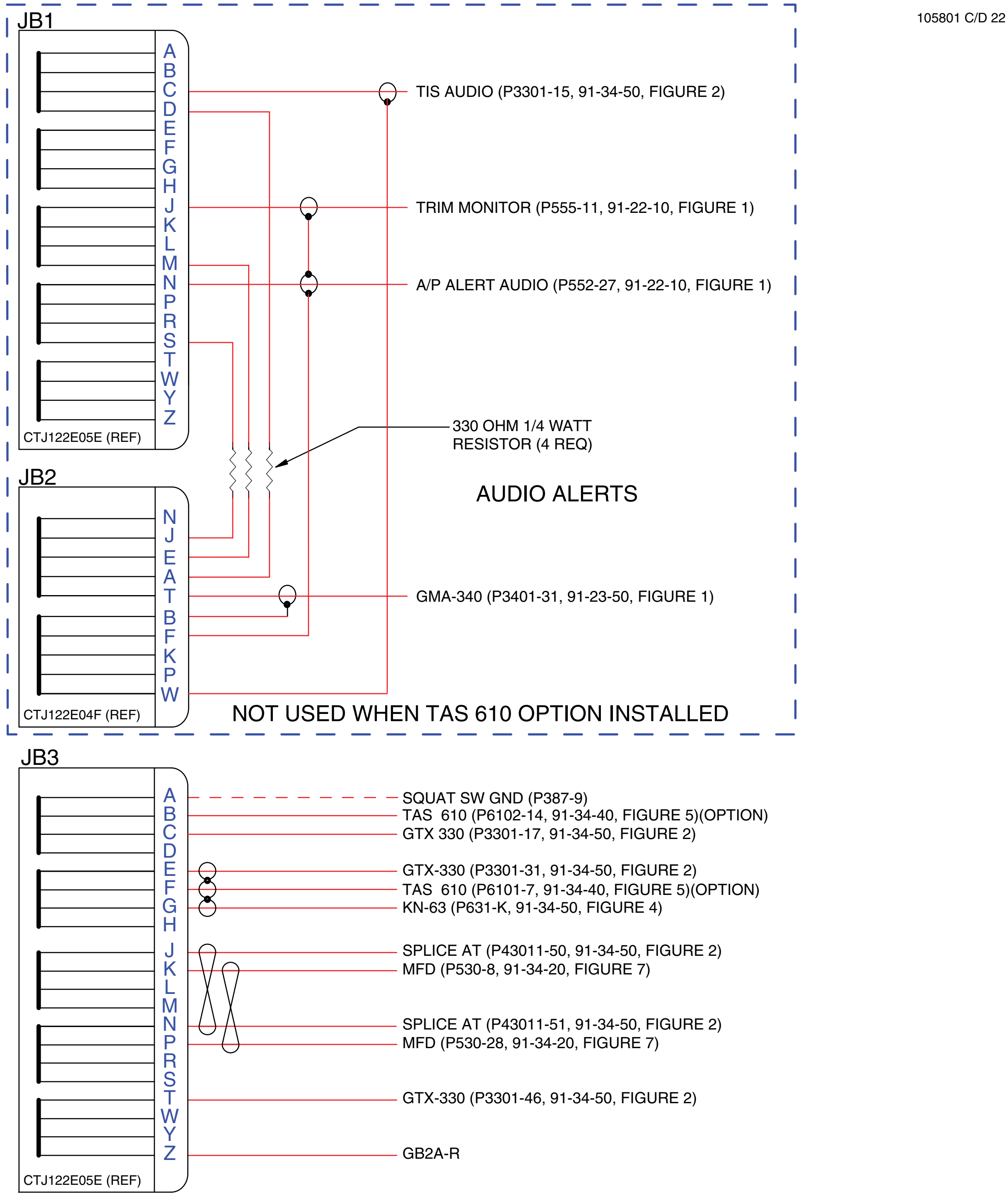
Junction Blocks
Figure 4 (Sheet 1 of 6)



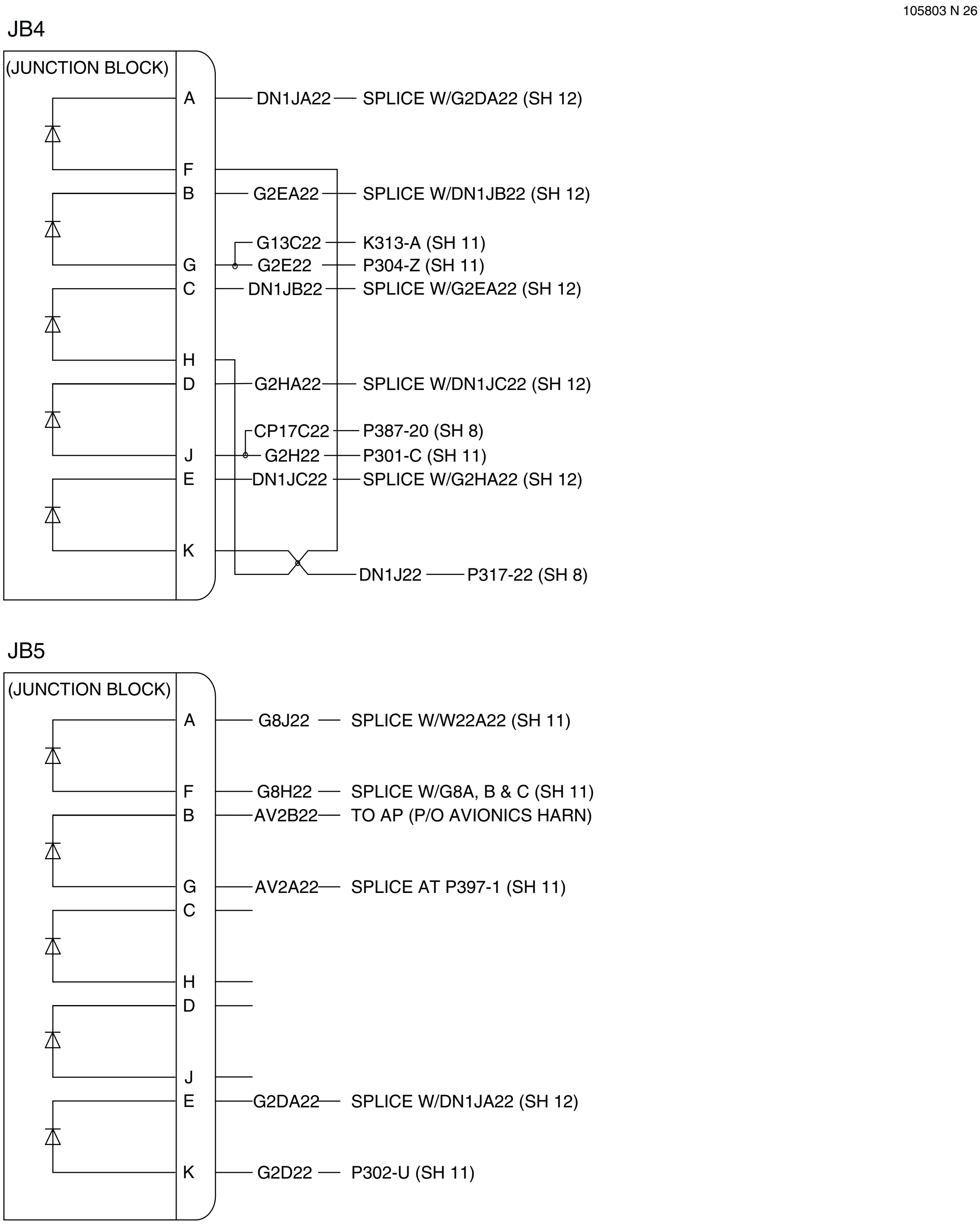
Junction Blocks
Figure 4 (Sheet 2 of 6)



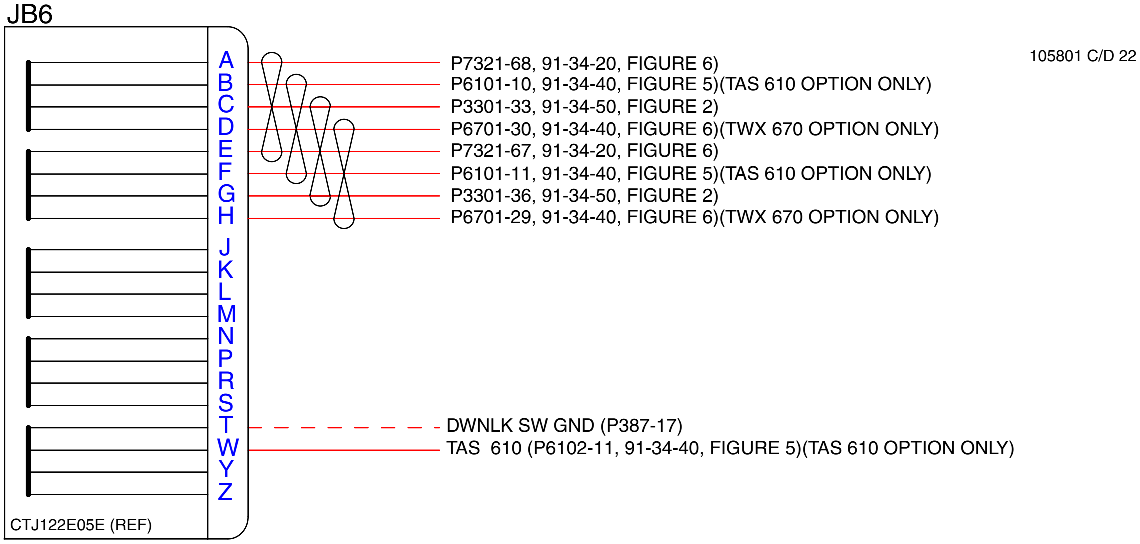
Junction Blocks
Figure 4 (Sheet 3 of 6)

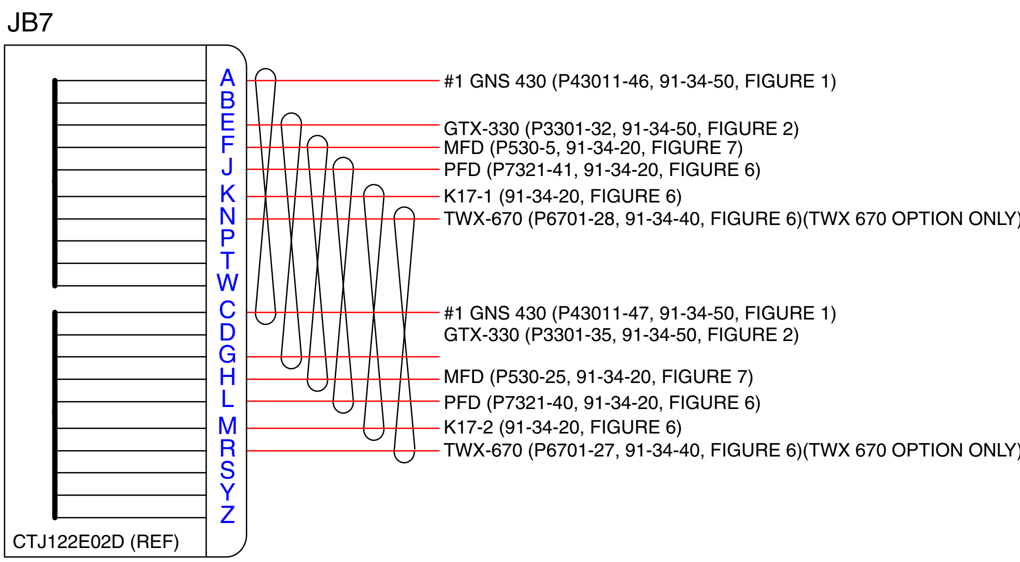


Junction Blocks
Figure 4 (Sheet 4 of 6)



Junction Blocks
Figure 4 (Sheet 5 of 6)

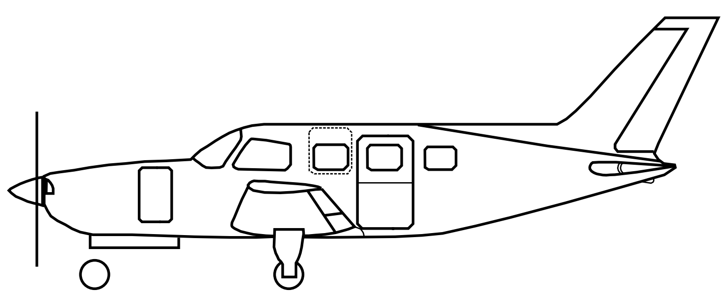




Junction Blocks
Figure 4 (Sheet 6 of 6)

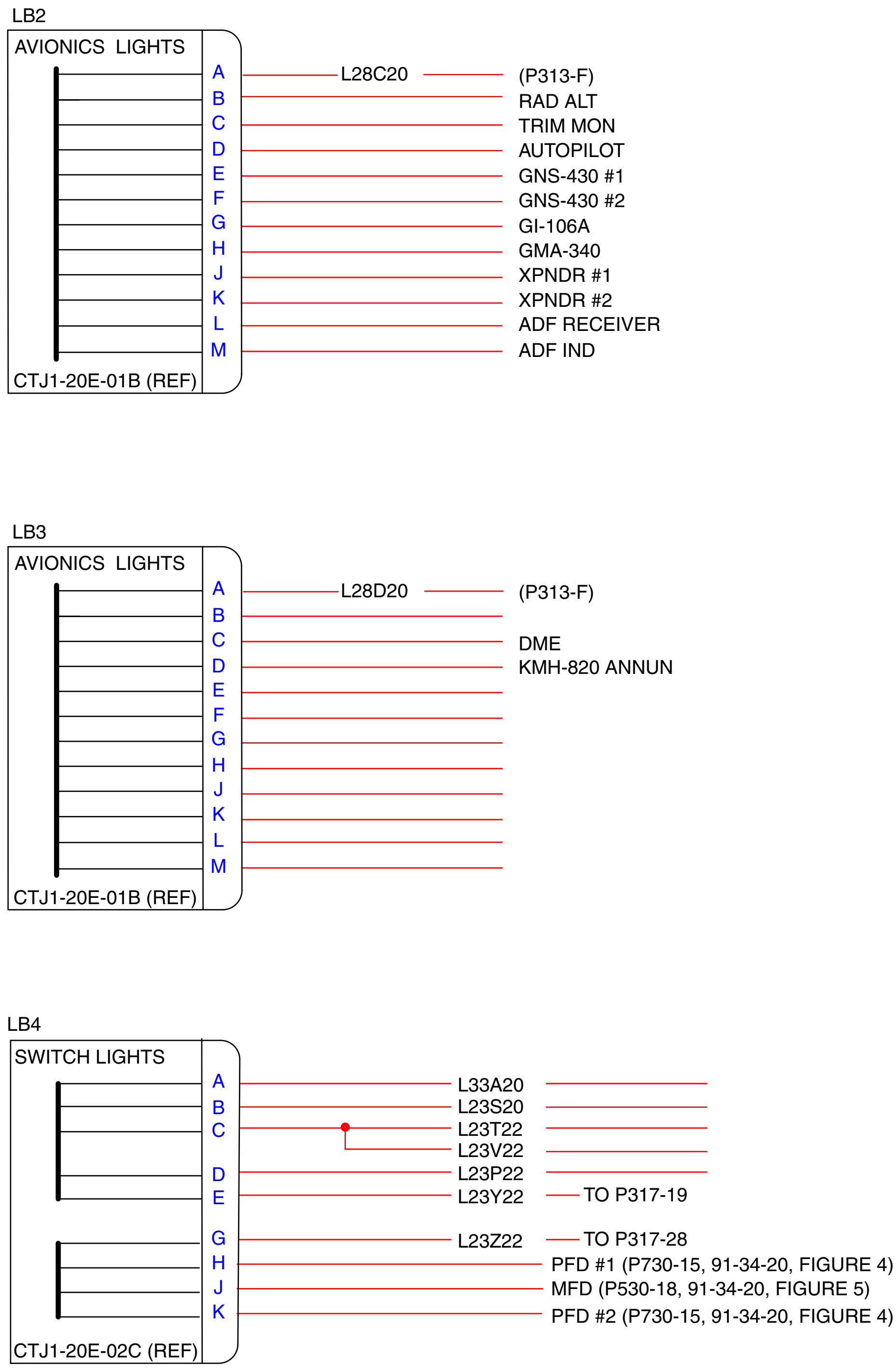
Item #	Designation	Description

Information Pending - See Parts Catalog

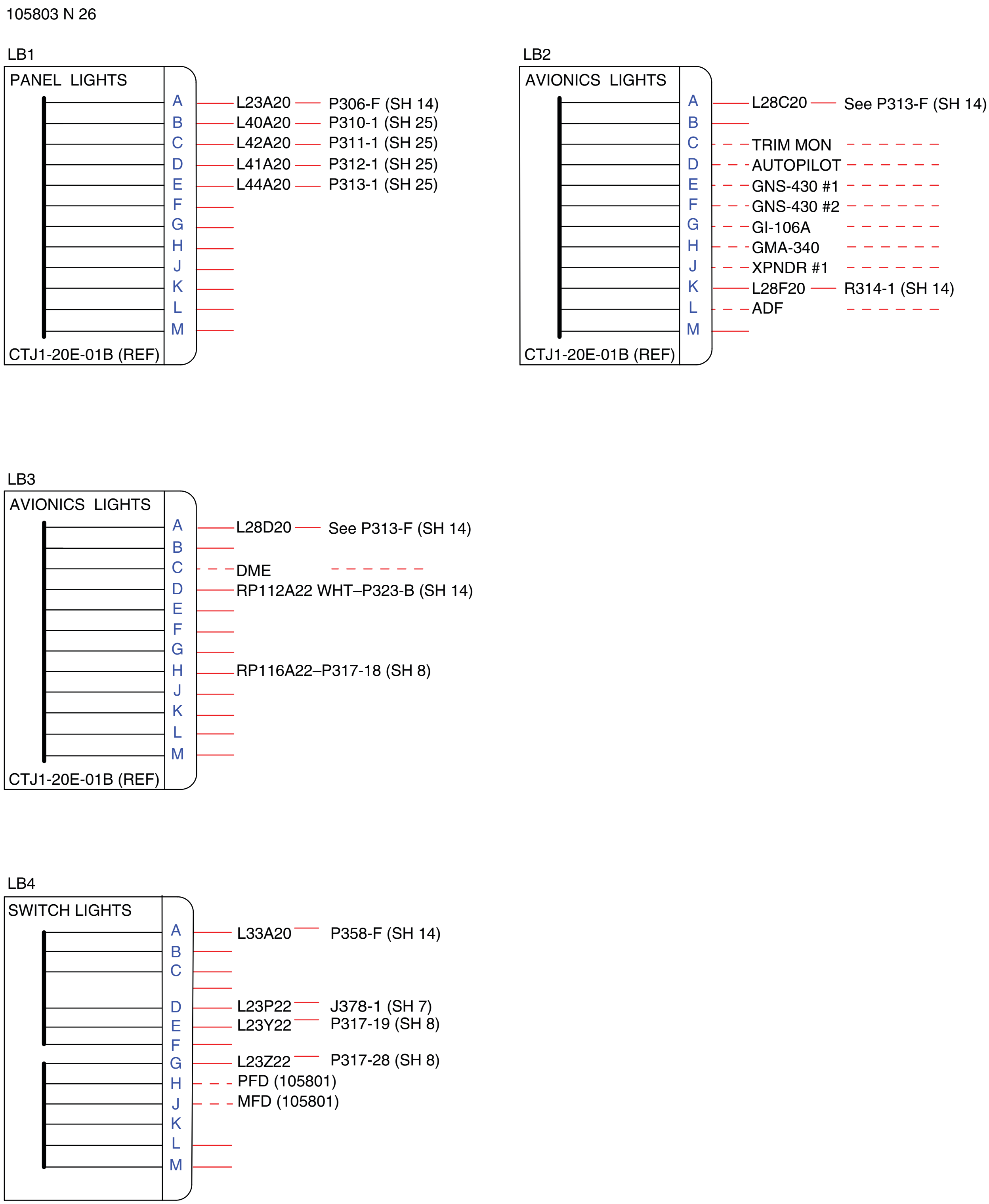


Lighting Blocks
Figure 5 (Sheet 1 of 3)

105130 A/F 30



Lighting Blocks
Figure 5 (Sheet 2 of 3)



Lighting Blocks
Figure 5 (Sheet 3 of 3)

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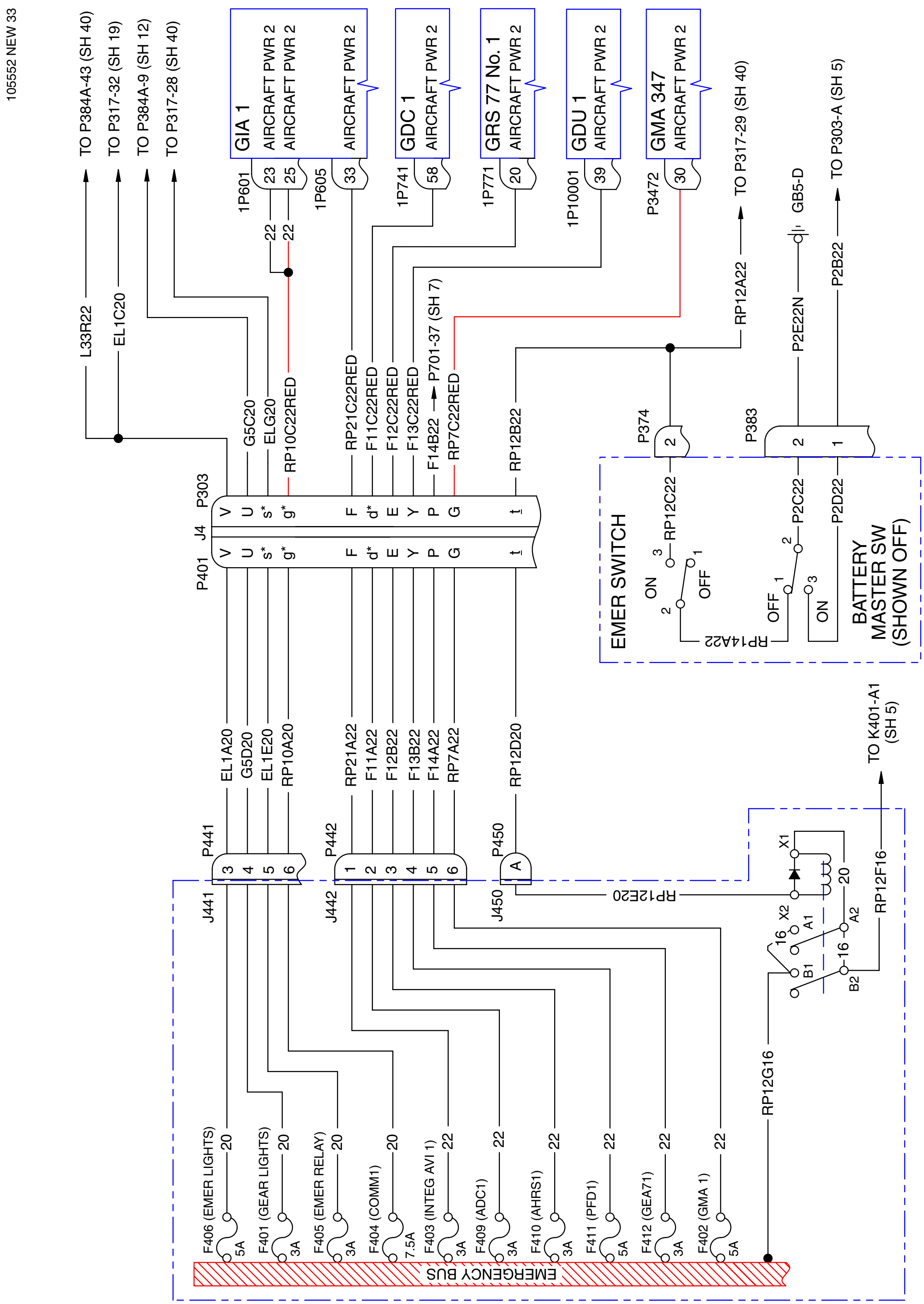
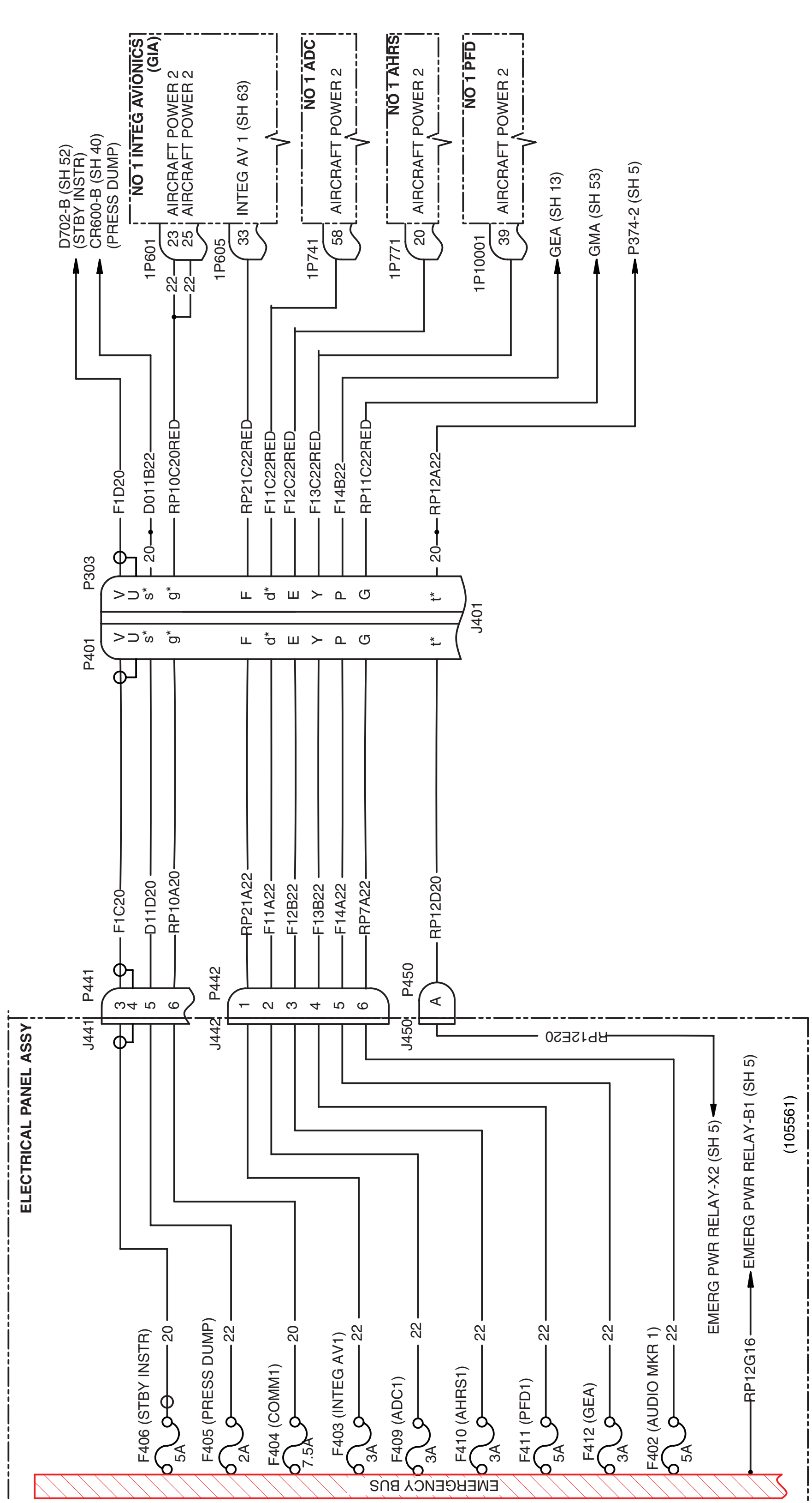
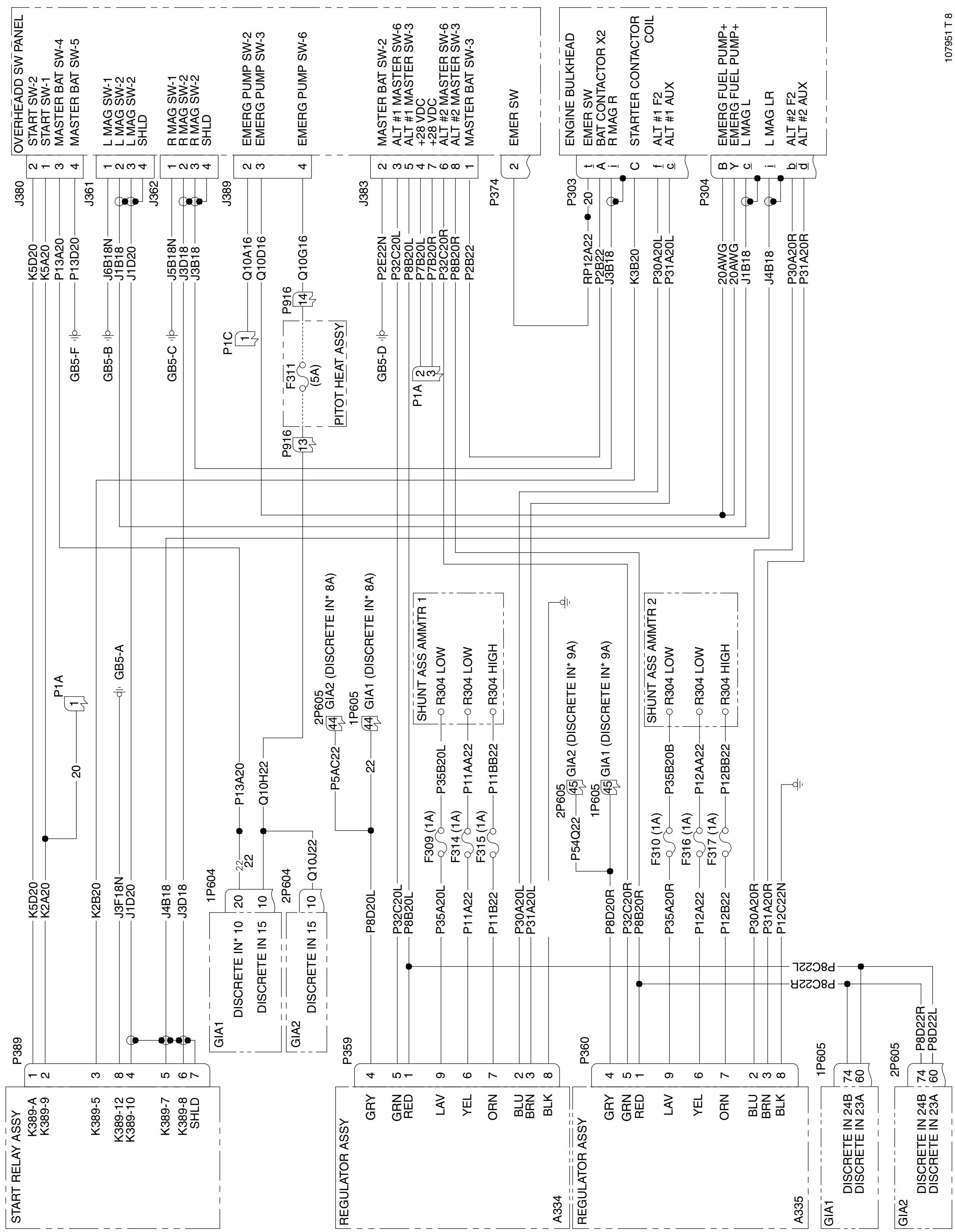


Emergency Power
Figure 6 (Sheet 1 of 2)

107975 NEW 7



Emergency Power
Figure 6 (Sheet 2 of 2)



Overhead Switch Panel
Figure 7

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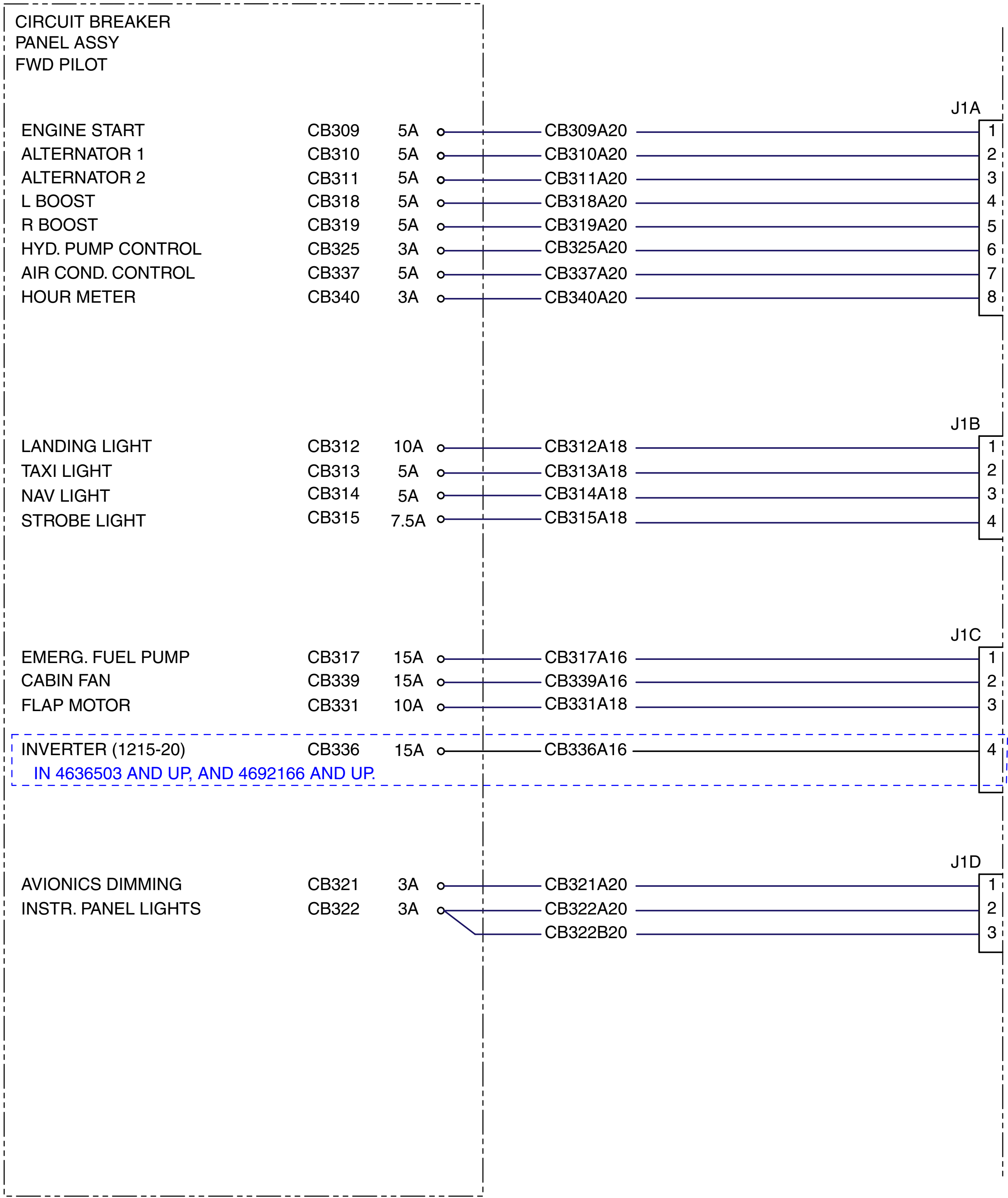
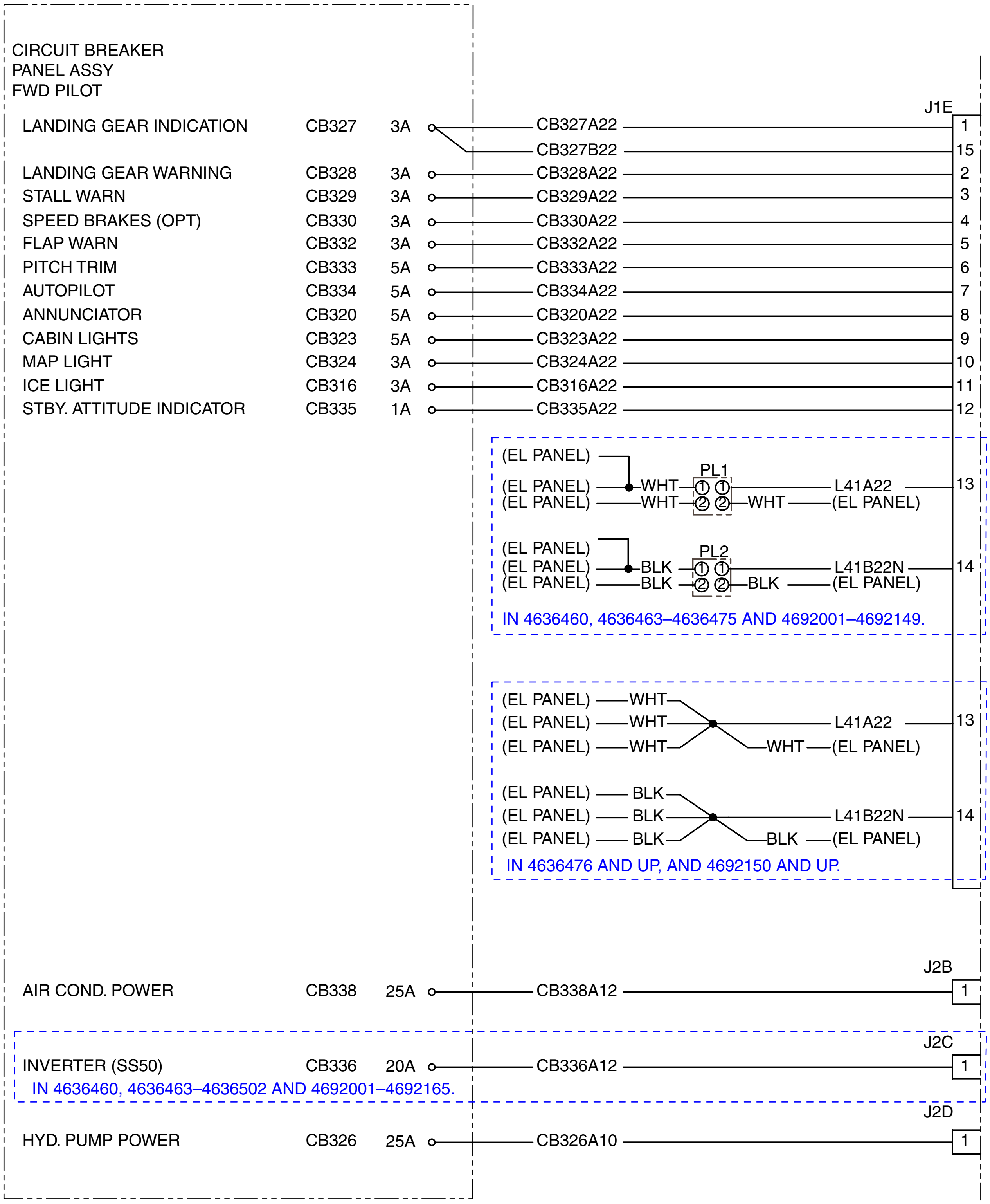


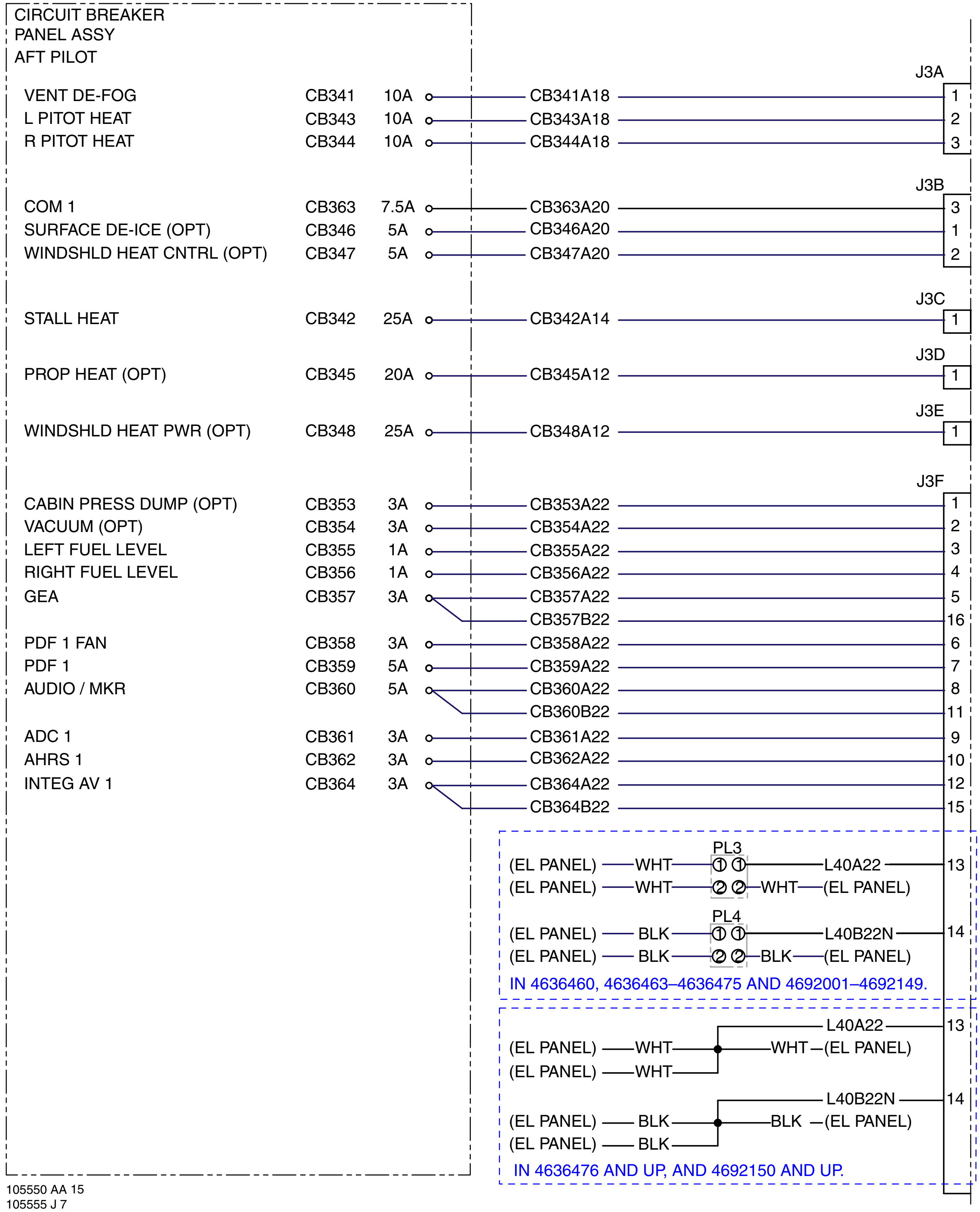
105555 H 4
105555 H 11

Circuit Breakers
Figure 1 (Sheet 1 of 4)

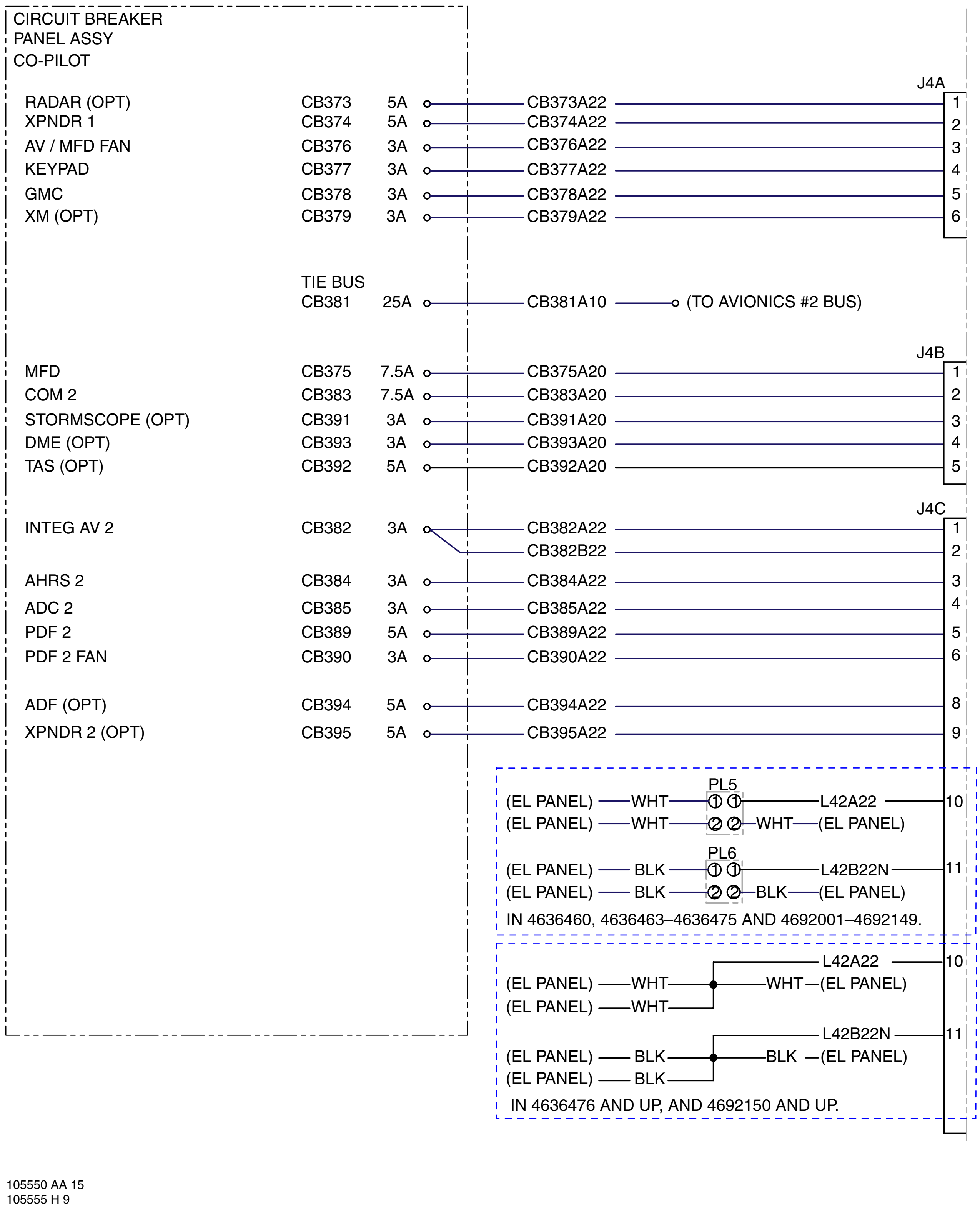


105550 AA 15
105555 H 5
105555 H 12

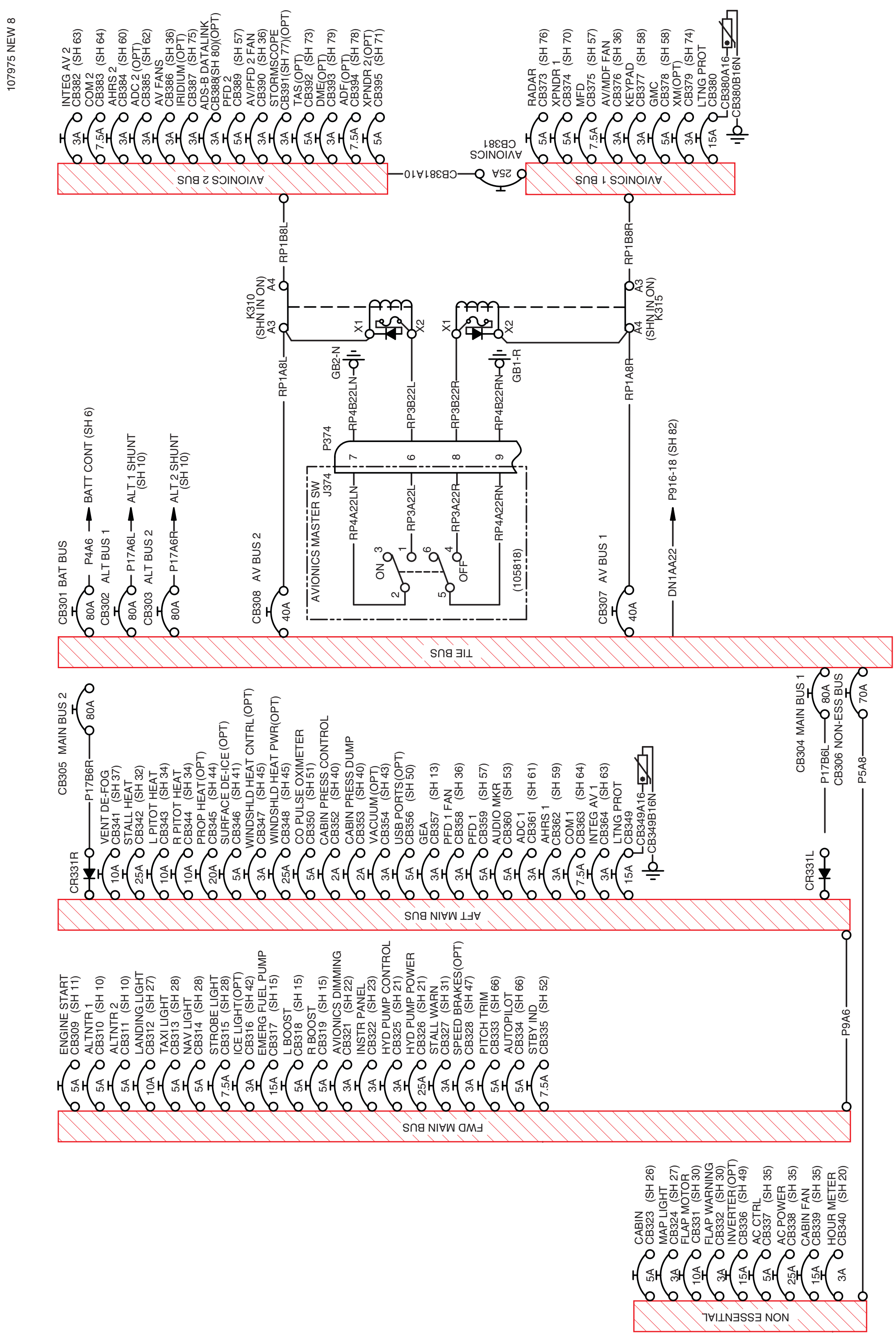
Circuit Breakers
Figure 1 (Sheet 2 of 4)



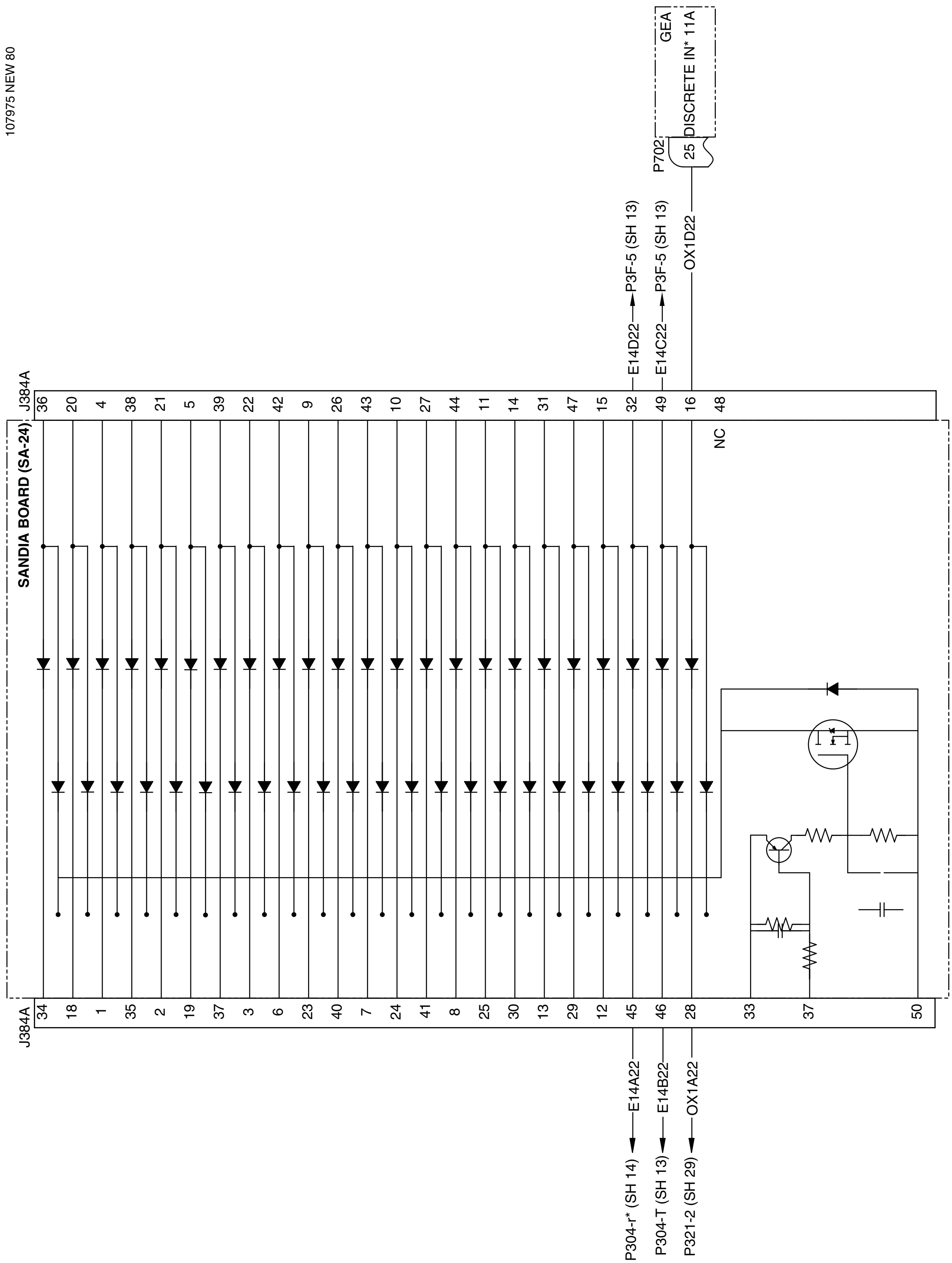
Circuit Breakers
Figure 1 (Sheet 3 of 4)



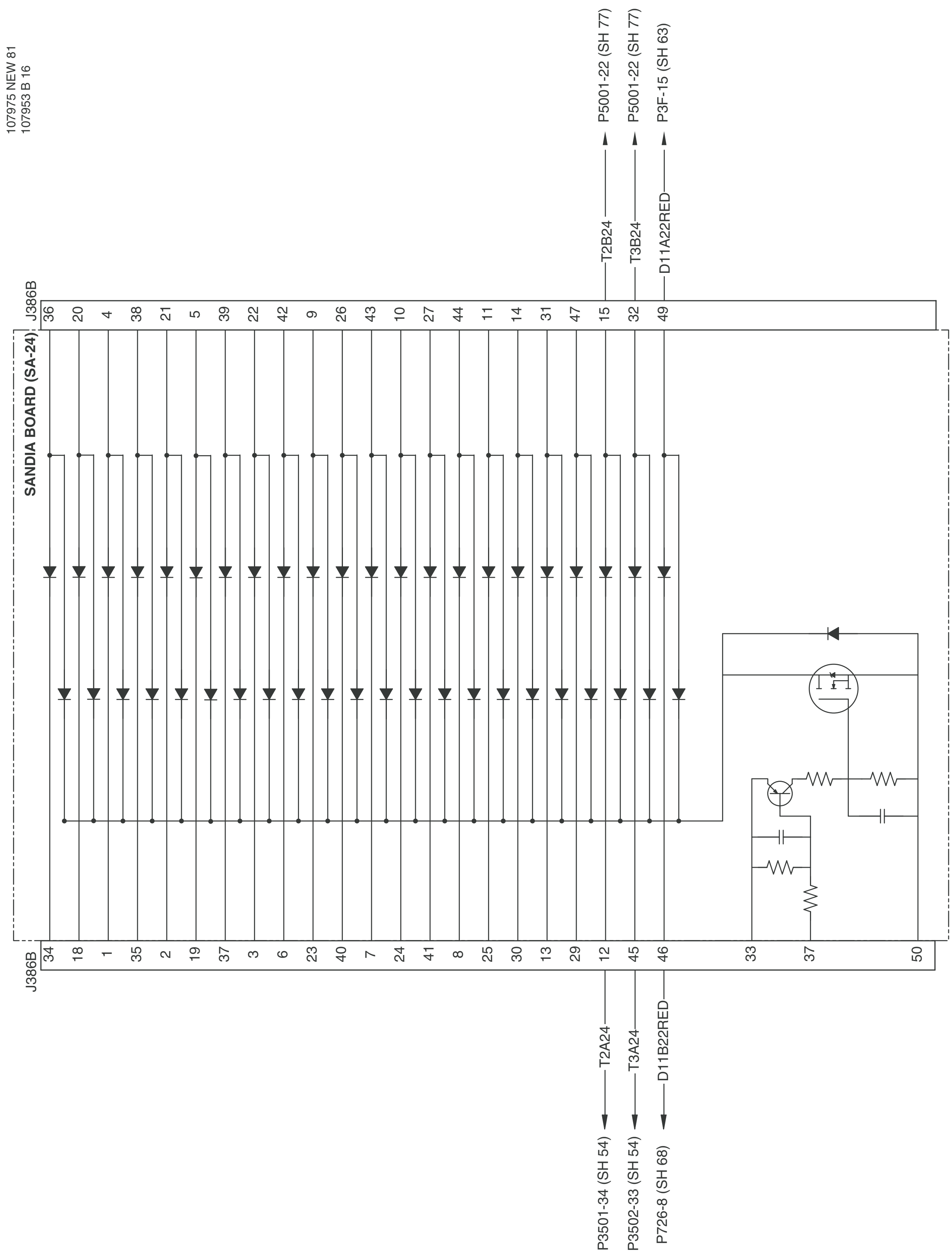
Circuit Breakers
Figure 1 (Sheet 4 of 4)



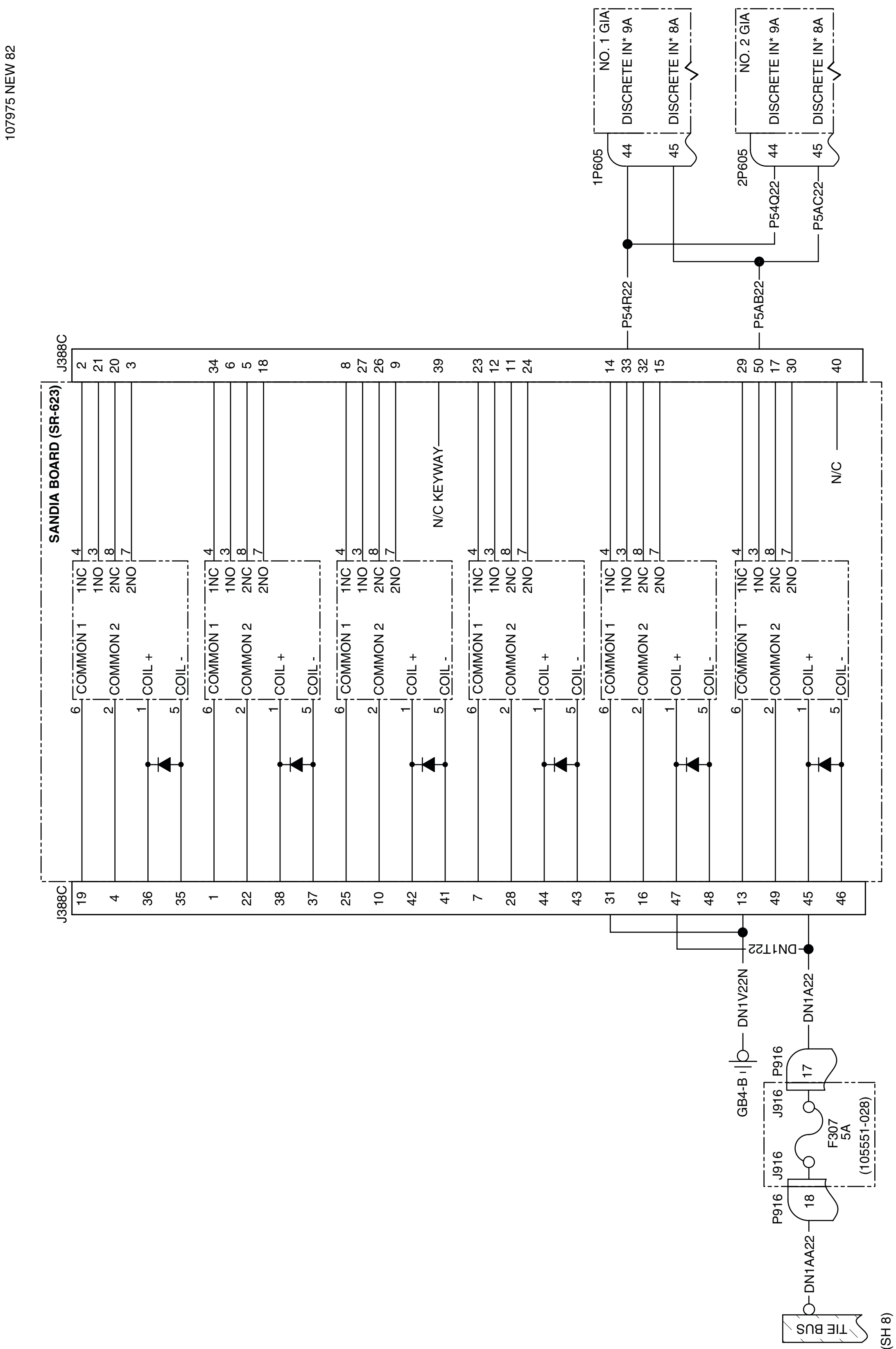
Power Distribution
Figure 2



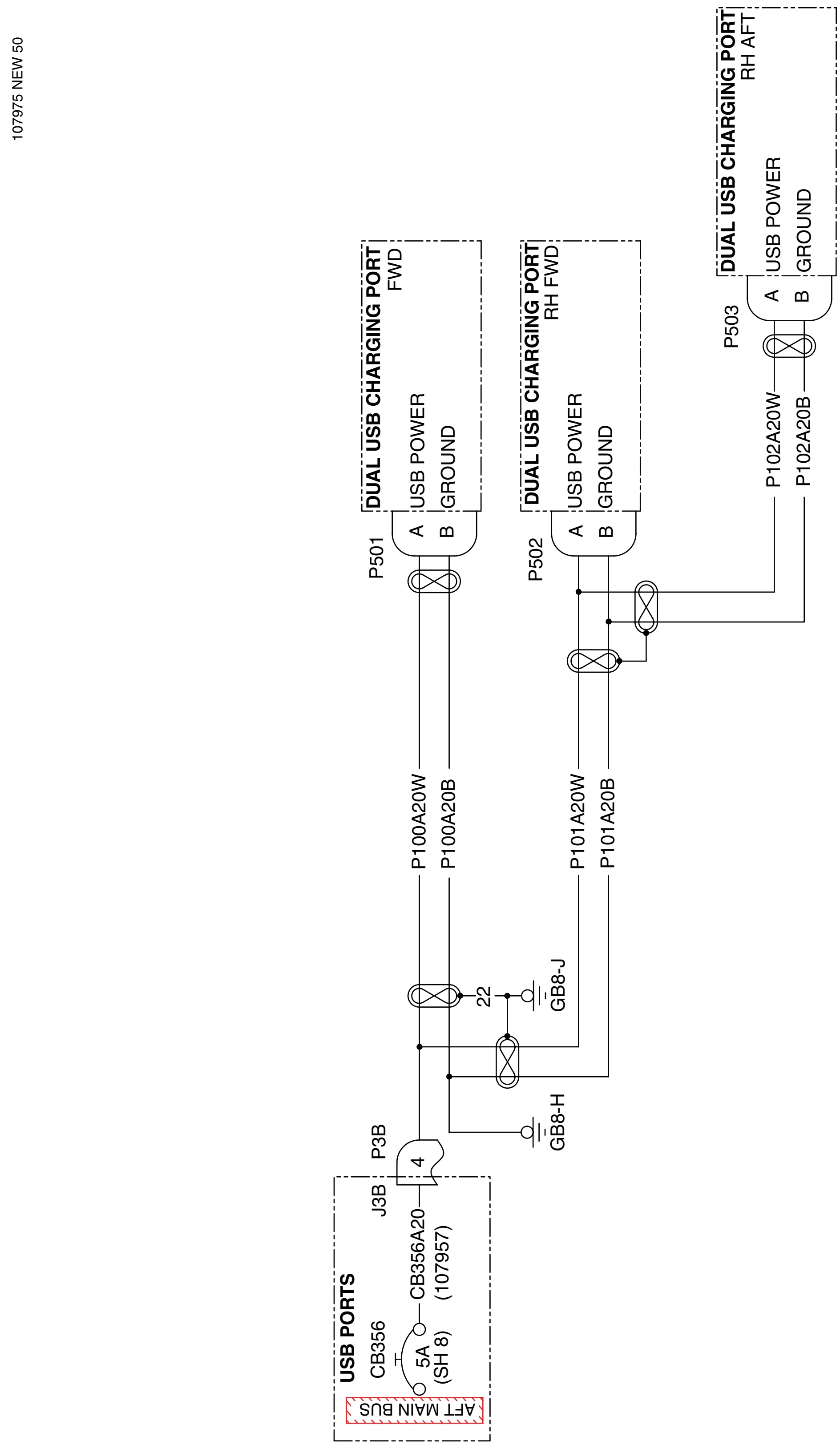
Diode Board
Figure 3 (Sheet 1 of 2)



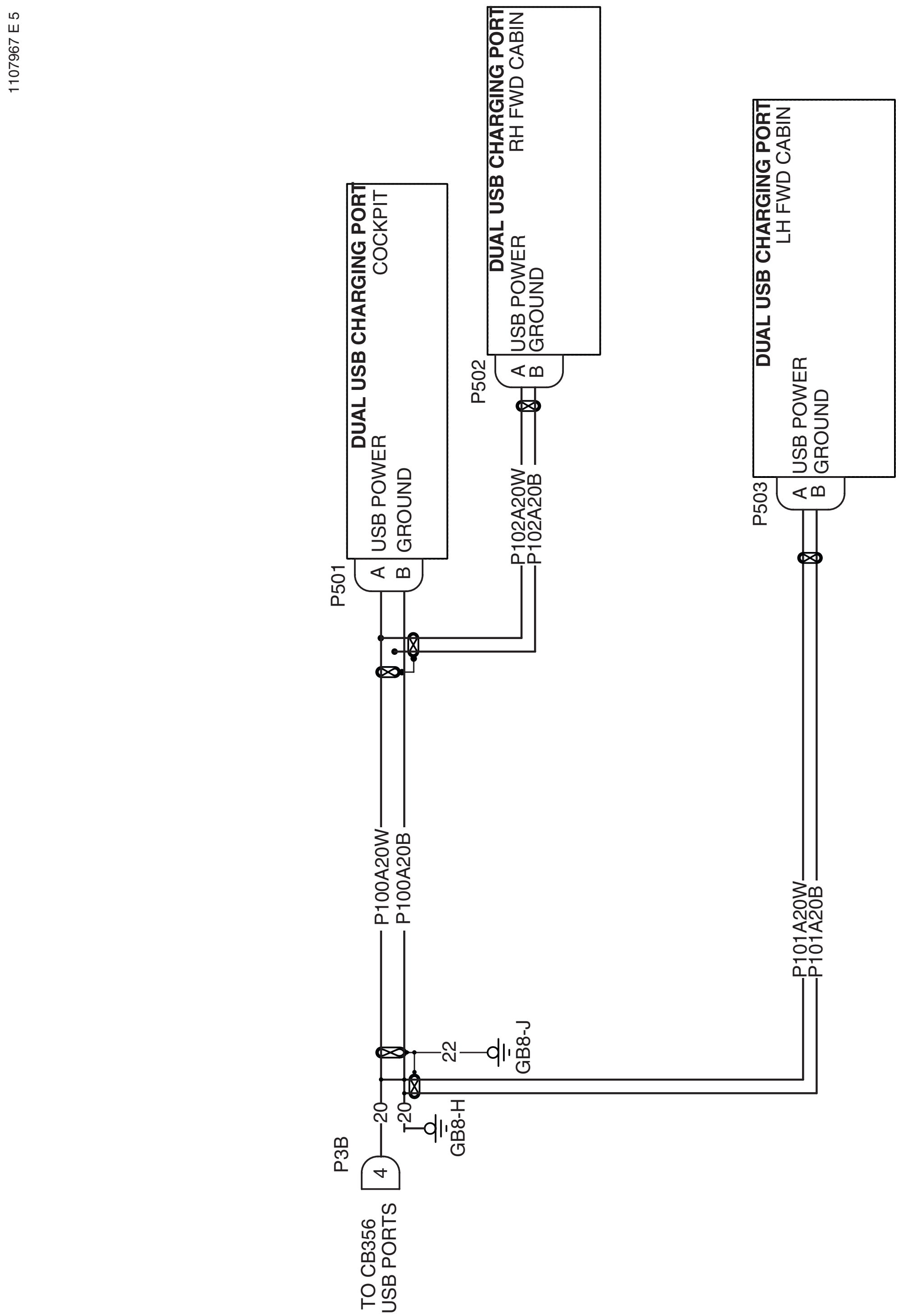
Diode Board
Figure 3 (Sheet 2 of 2)



Relay Board
Figure 4



USB Charging Ports (Optional)
Figure 5 (Sheet 1 of 2)



USB Charging Ports (Optional)
Figure 5 (Sheet 2 of 2)

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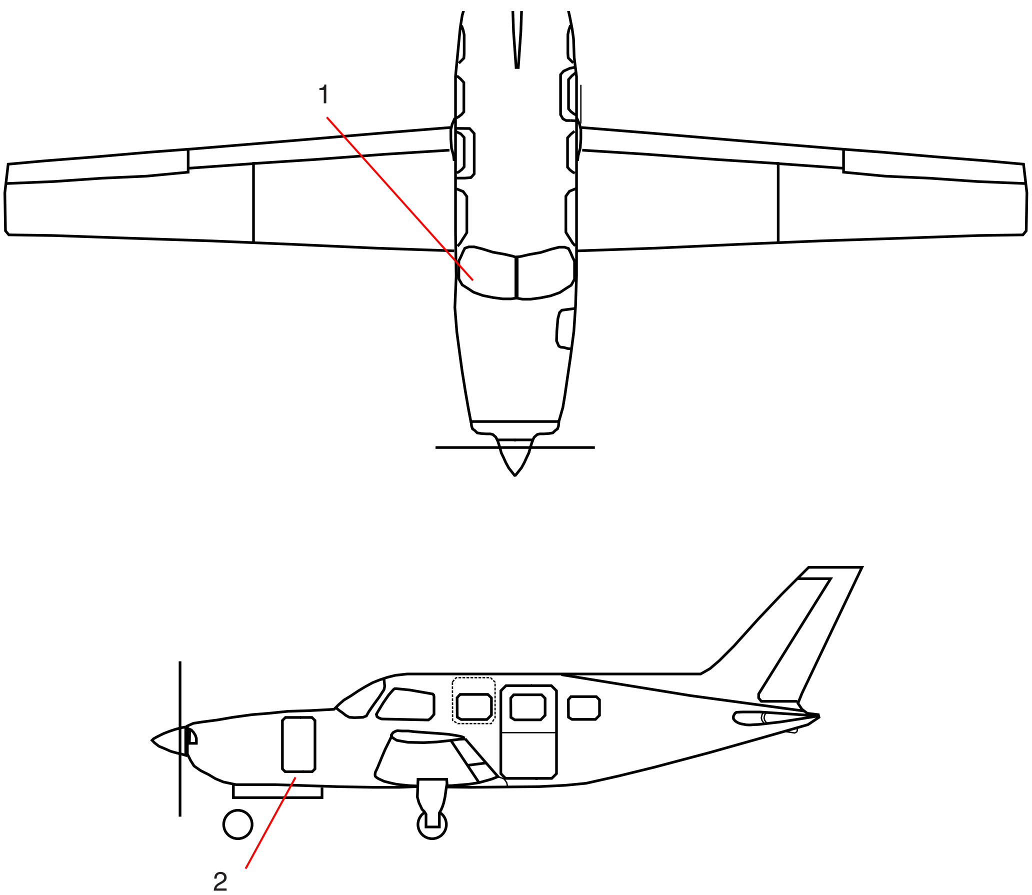
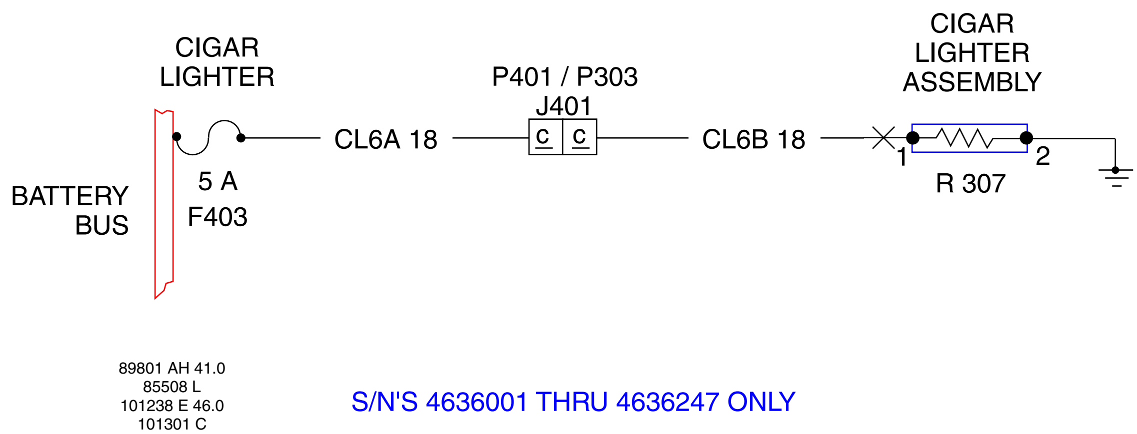
Item #	Designation	Description
1	R307	Cigar Lighter (1)
		or
	R307	Power Point (2)
2	F403	Fuse (5 Amp)

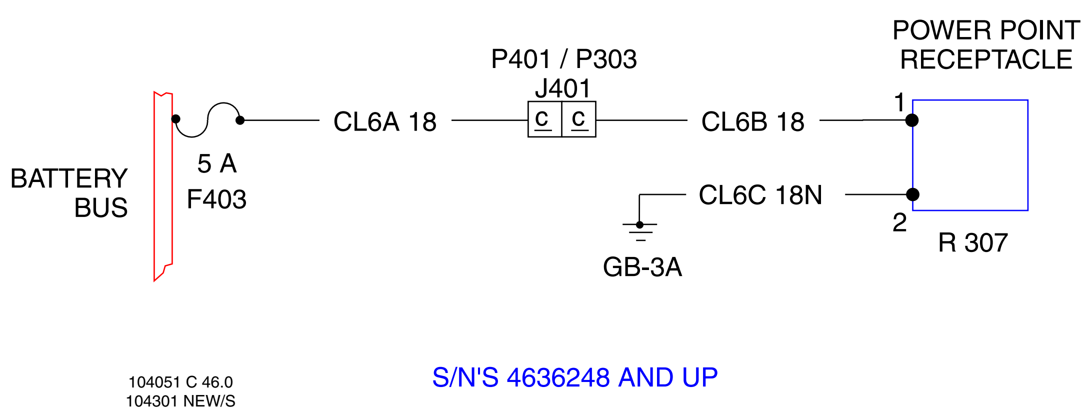
(1) - 4636001 thru 46360247 only
(2) - 4636248 and up



Cigar Lighter / Power Point
Figure 1 (Sheet 1 of 2)



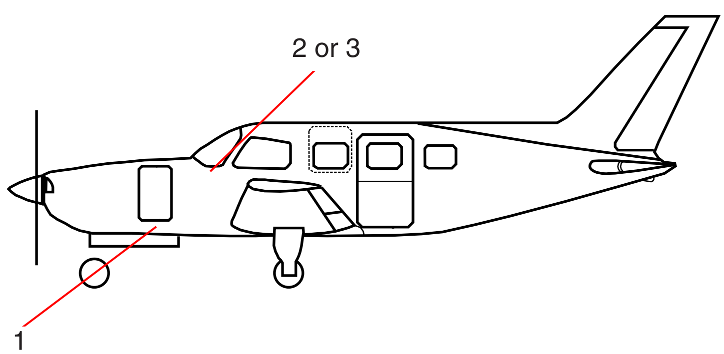
S/N'S 4636001 THRU 4636247 ONLY



S/N'S 4636248 AND UP

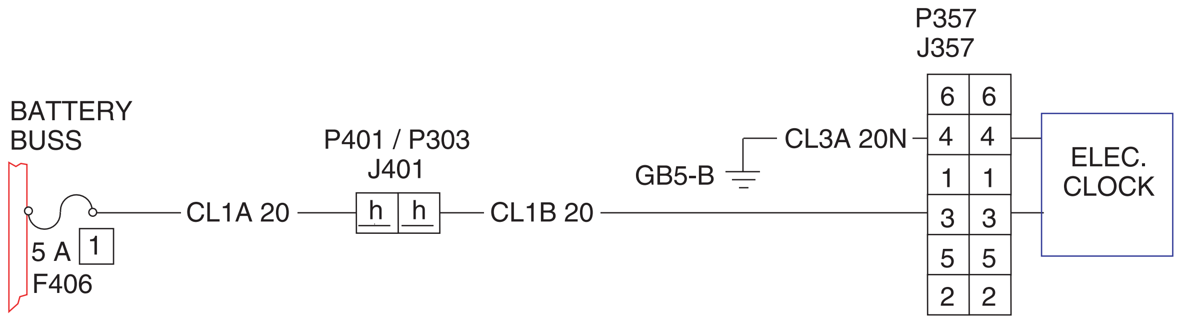
Cigar Lighter / Power Point
Figure 1 (Sheet 2 of 2)

Item #	Designation	Description
1	F406	Fuse (5 Amp)
2	LC-2	Digital Quartz Clock
3	LC-6	Digital Quartz Clock



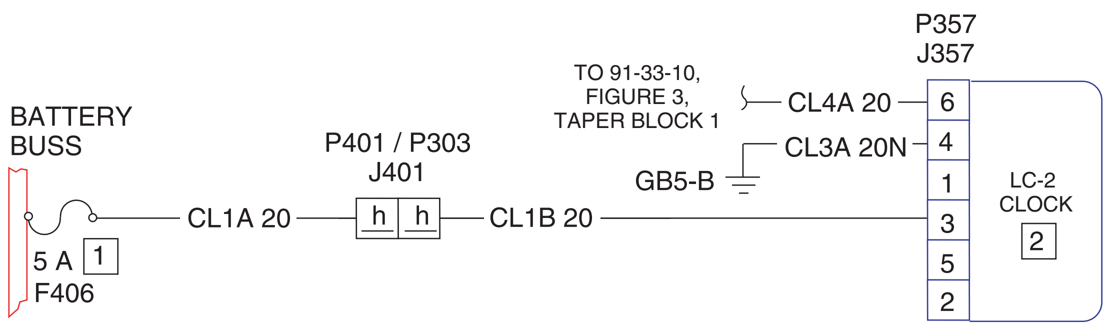
Electric and Digital Clocks
Figure 2 (Sheet 1 of 2)

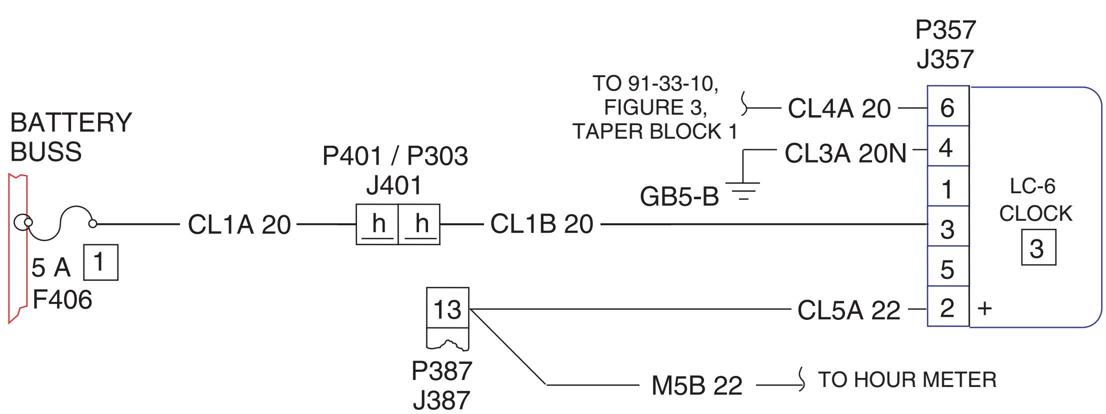
89801 AH 39.0/39.1



STANDARD

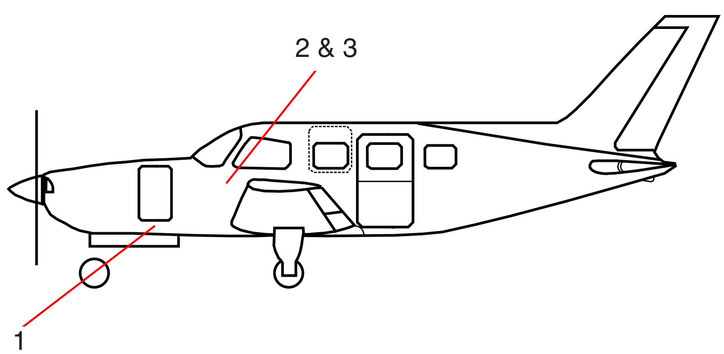
DIGITAL CLOCK (OPTIONAL)



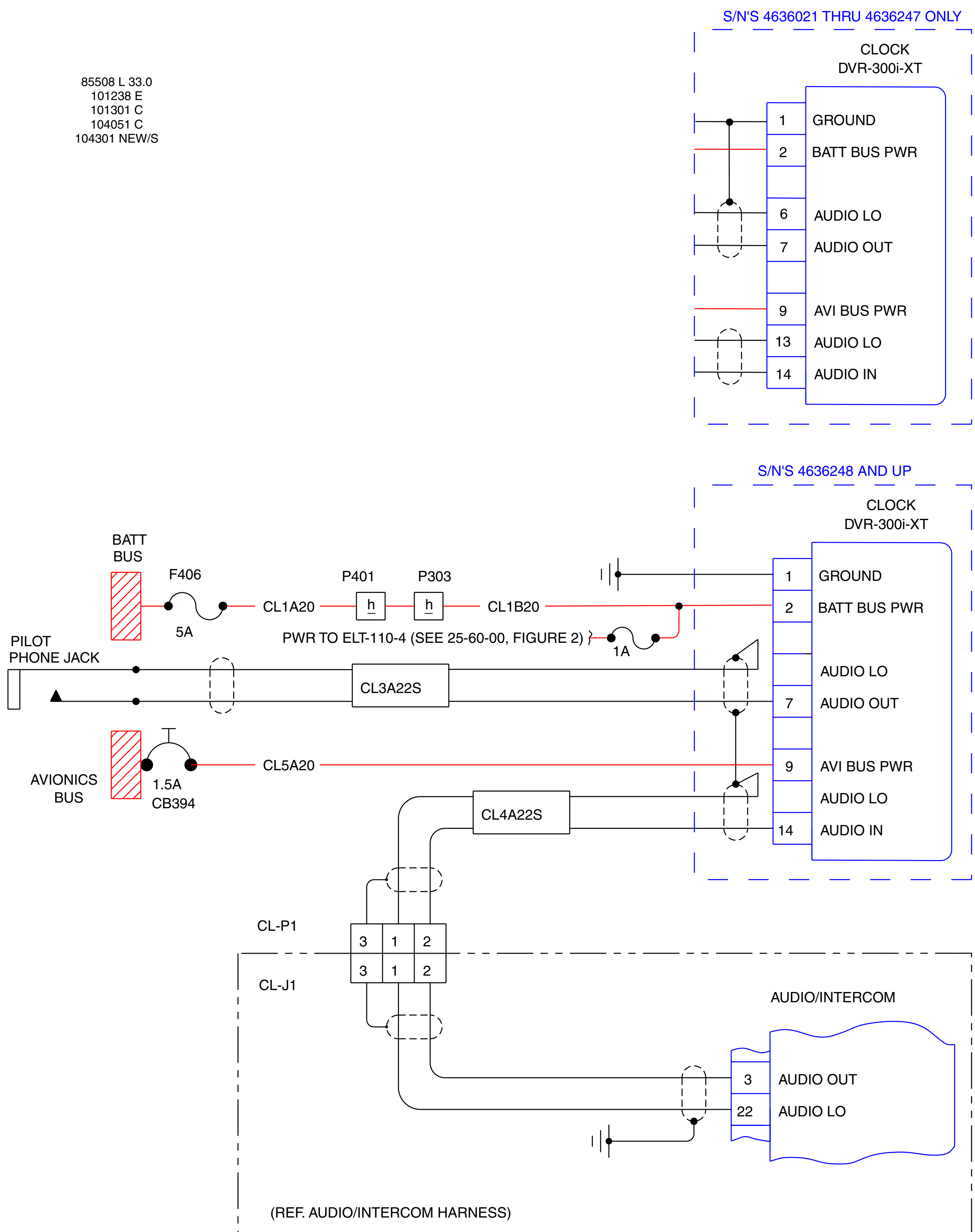


Electric and Digital Clocks
Figure 2 (Sheet 2 of 2)

Item #	Designation	Description
1	F406	Fuse (5 Amp)
2	CB394	Circuit Breaker (1.5 Amp)
3		Digital Quartz Clock

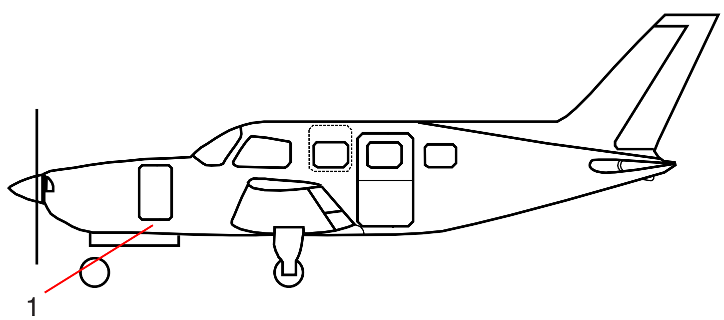


Digital Voice Recorder Clock
Figure 3 (Sheet 1 of 2)

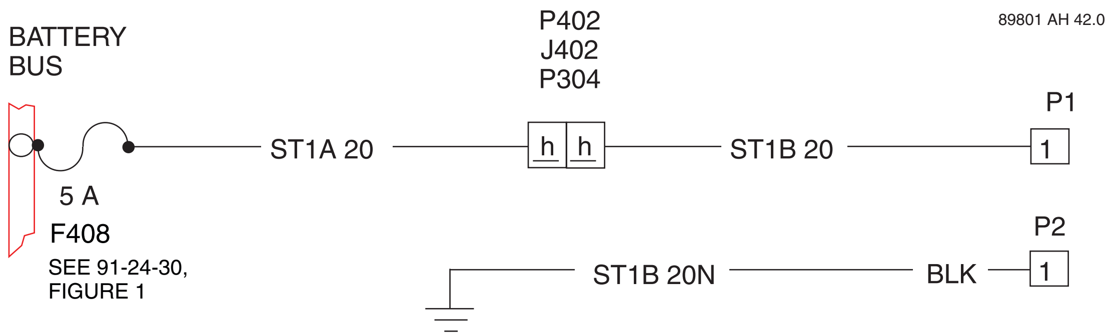


Digital Voice Recorder Clock
Figure 3 (Sheet 2 of 2)

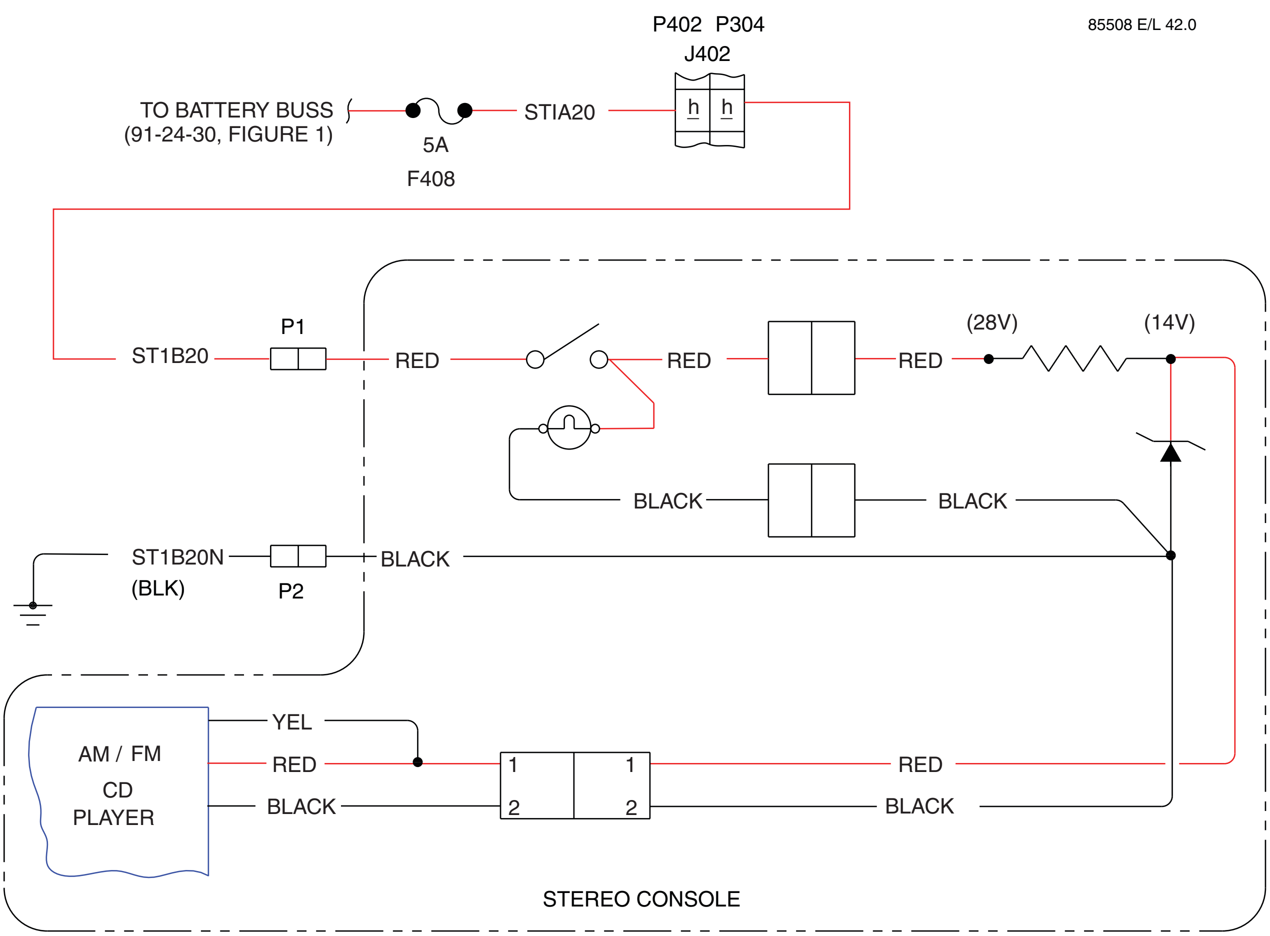
Item #	Designation	Description
1	F408	Fuse (5 Amp)



Stereo (Optional)
Figure 4 (Sheet 1 of 4)



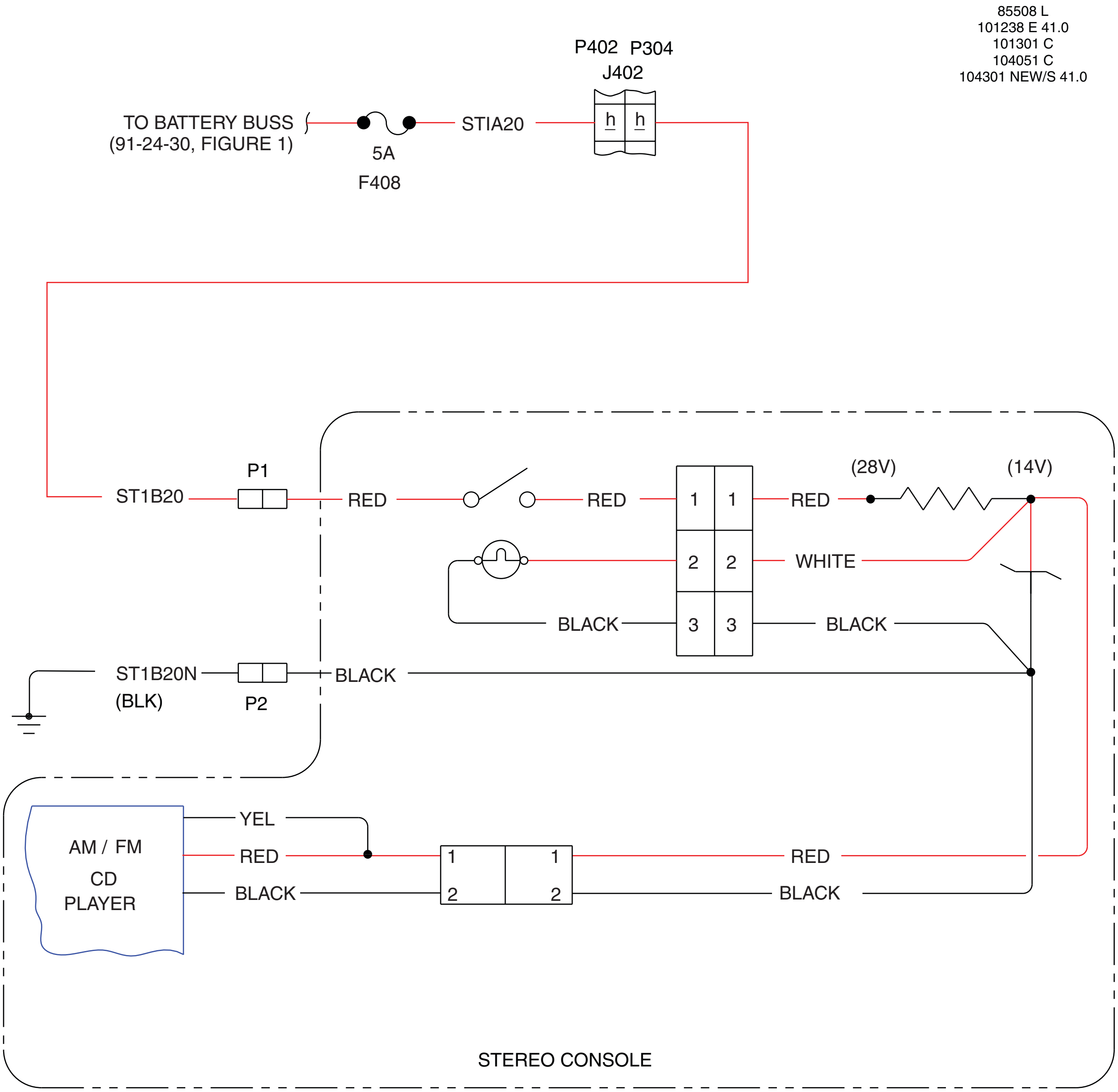
S/N'S 4636001 THRU 4636020 ONLY



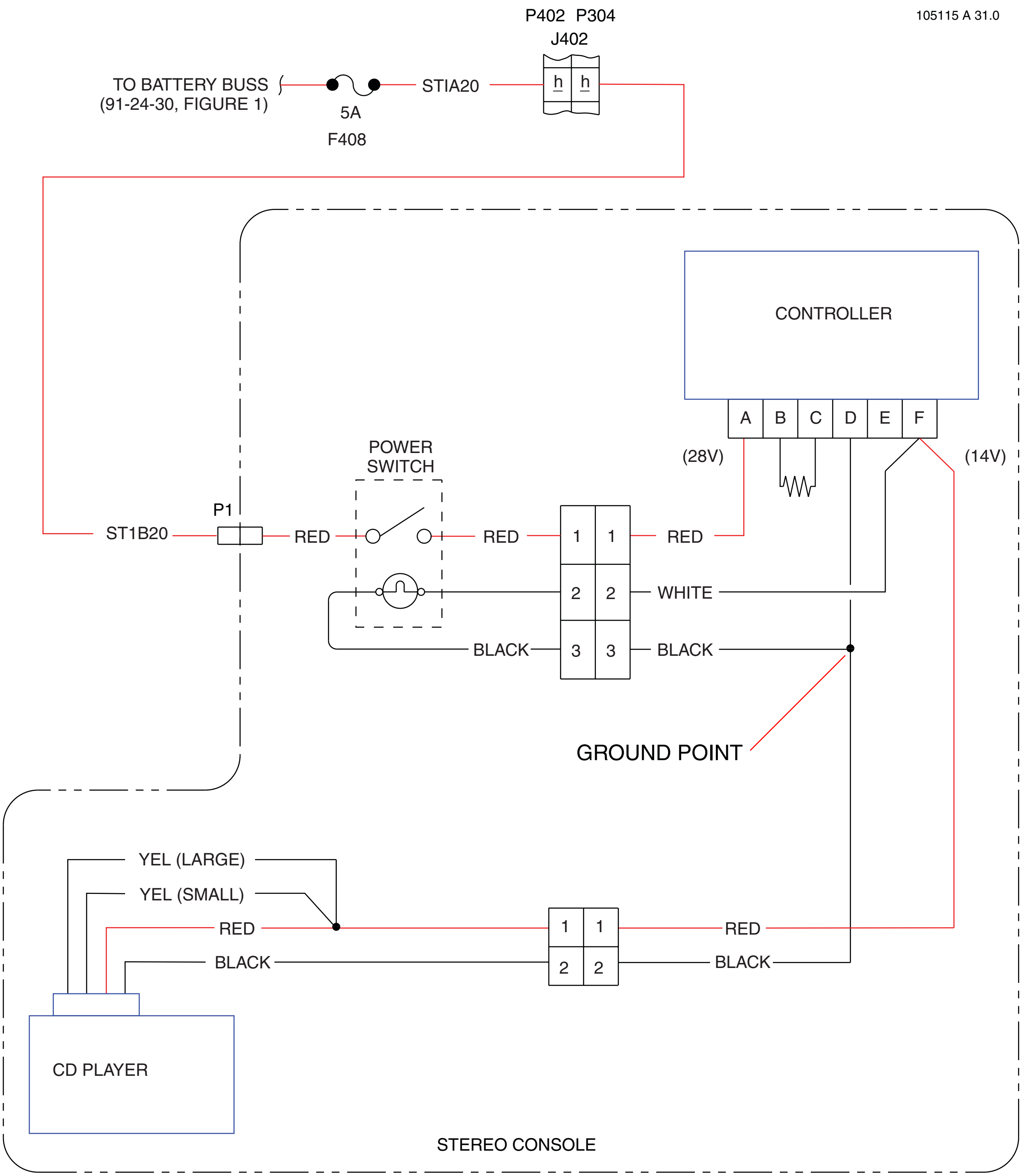
S/N'S 4636021 THRU 4636076 ONLY

Stereo (Optional)
Figure 4 (Sheet 2 of 4)

Effectivity
4636001 thru 4636076



Stereo (Optional)
Figure 4 (Sheet 3 of 4)



Stereo (Optional)
Figure 4 (Sheet 4 of 4)

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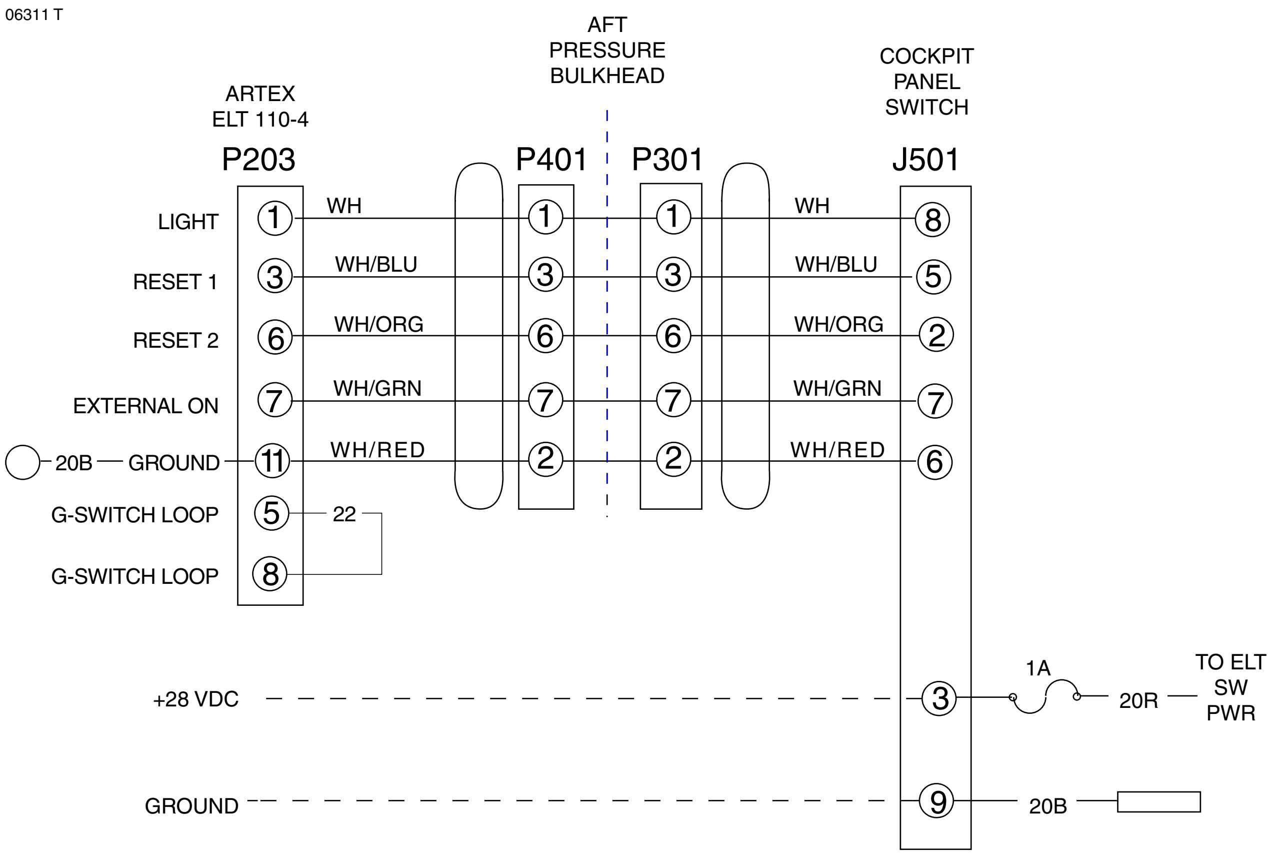
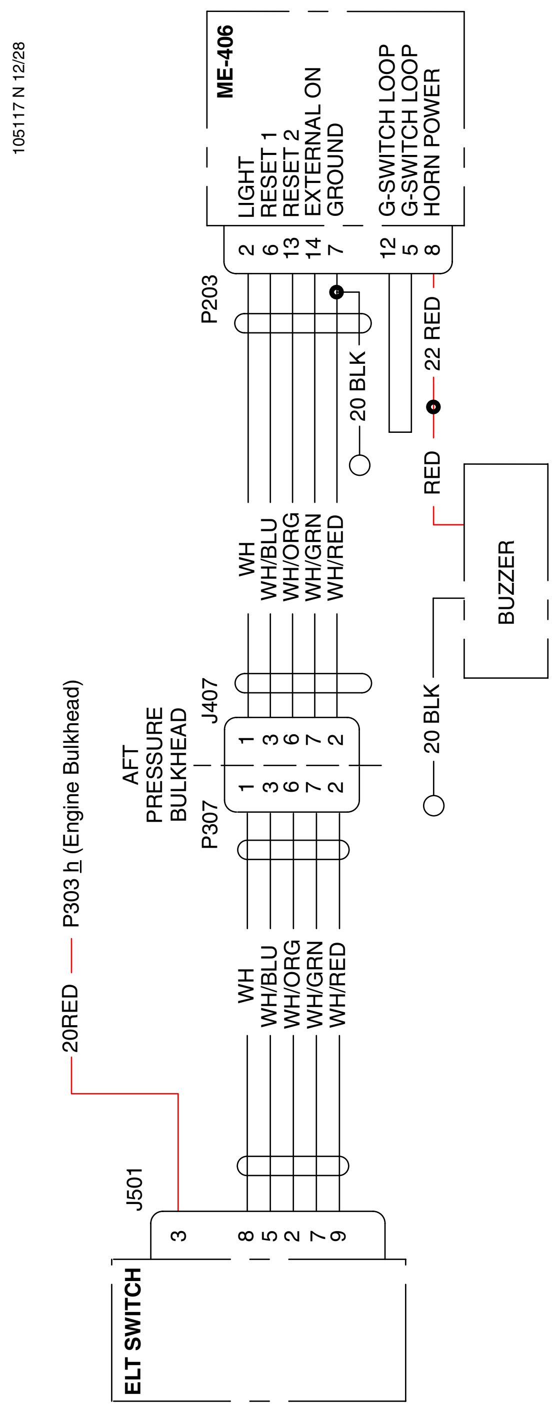
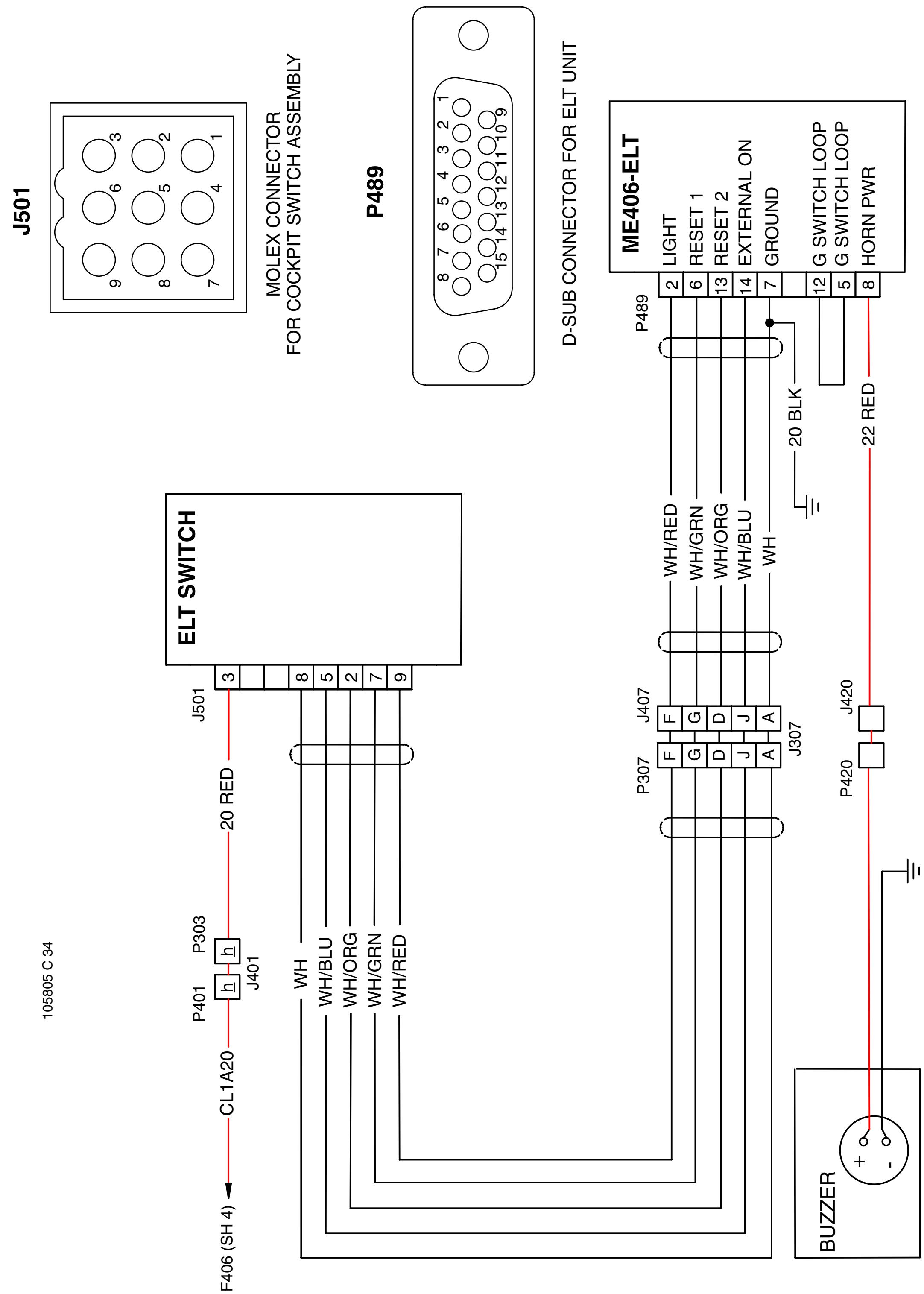


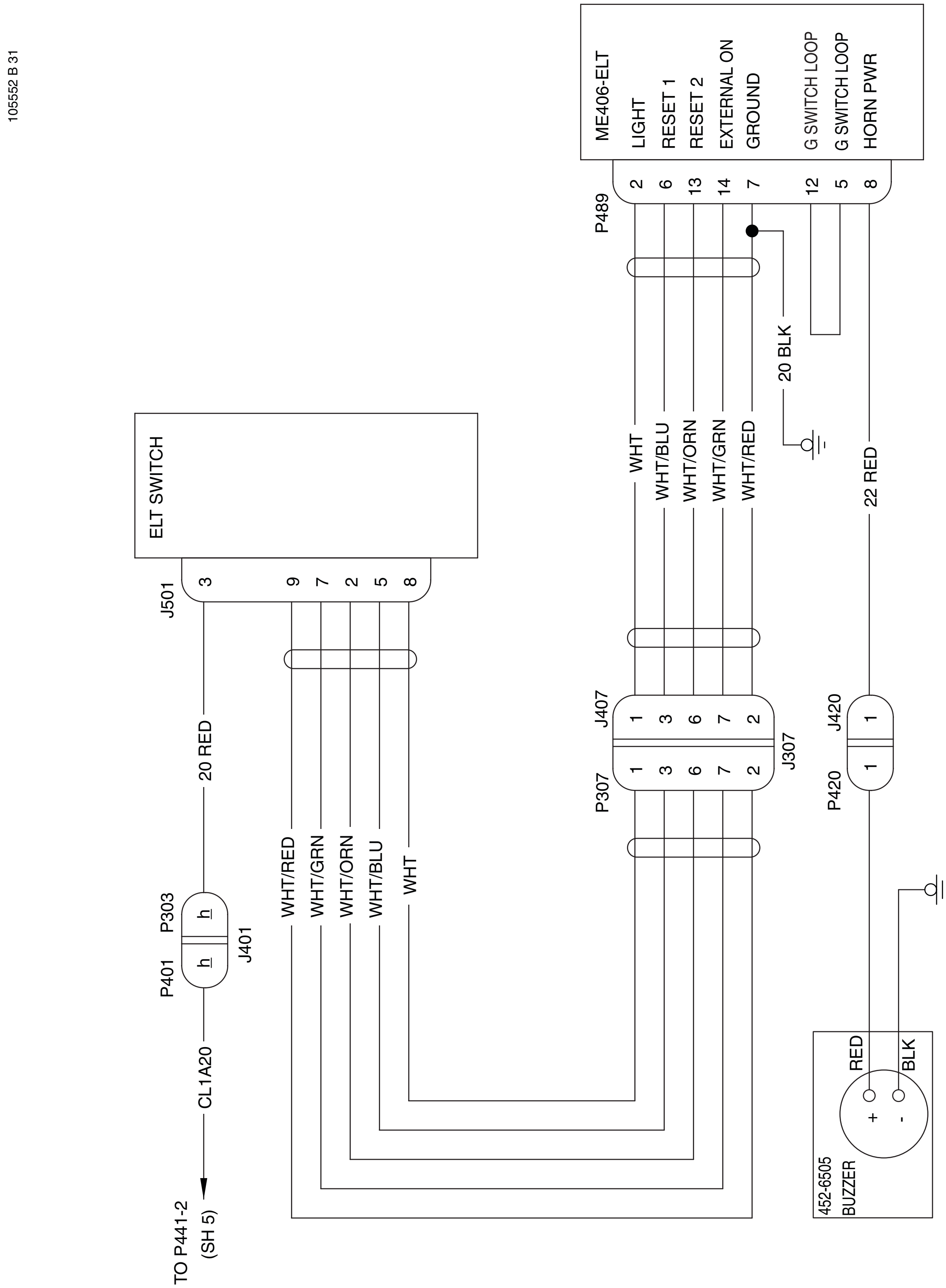
Artex 110 ELT
Figure 1



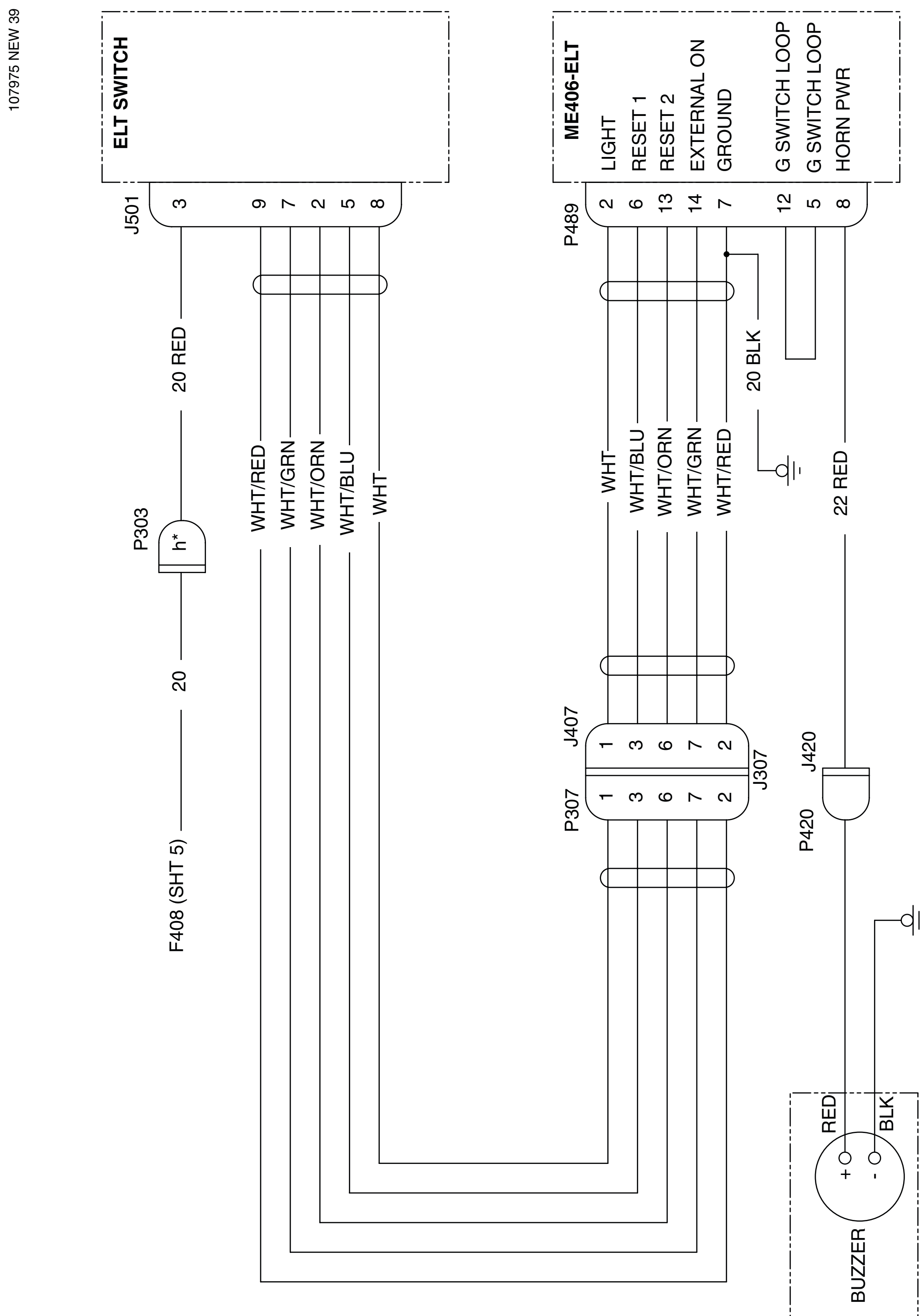
Artex ME406 ELT
Figure 2 (Sheet 1 of 4)



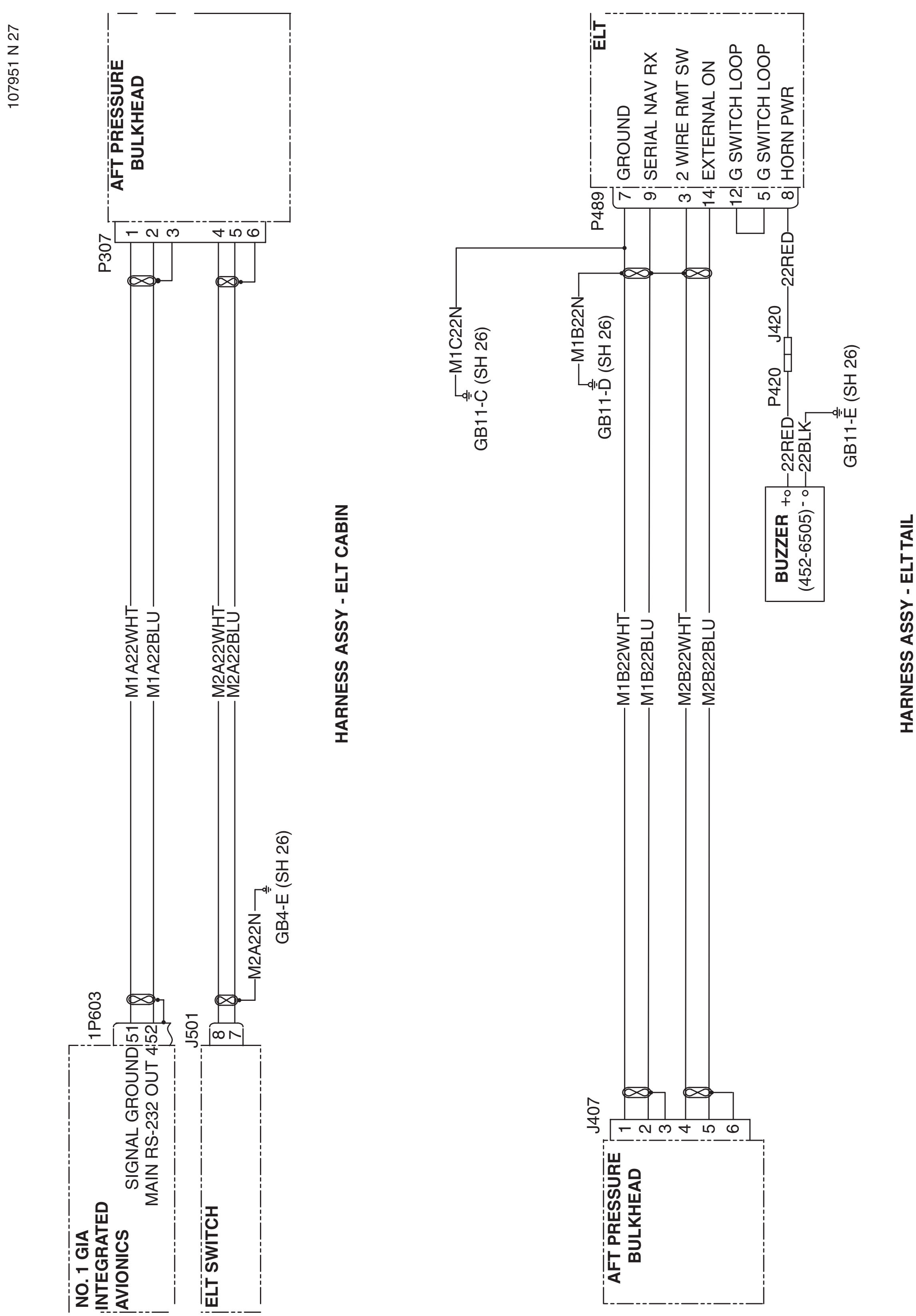
Artex ME406 ELT
Figure 2 (Sheet 2 of 4)



Artex ME406 ELT
Figure 2 (Sheet 3 of 4)



Artex ME406 ELT
Figure 2 (Sheet 4 of 4)



Artex ELT 1000
Figure 3

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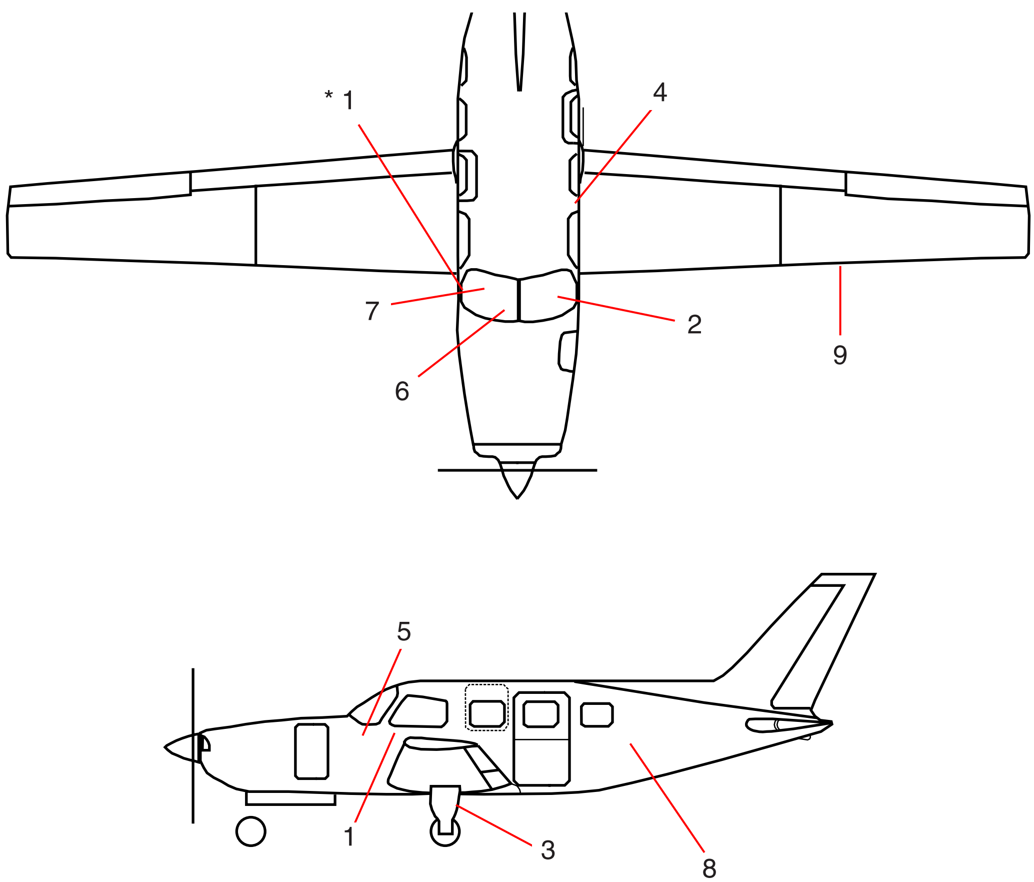
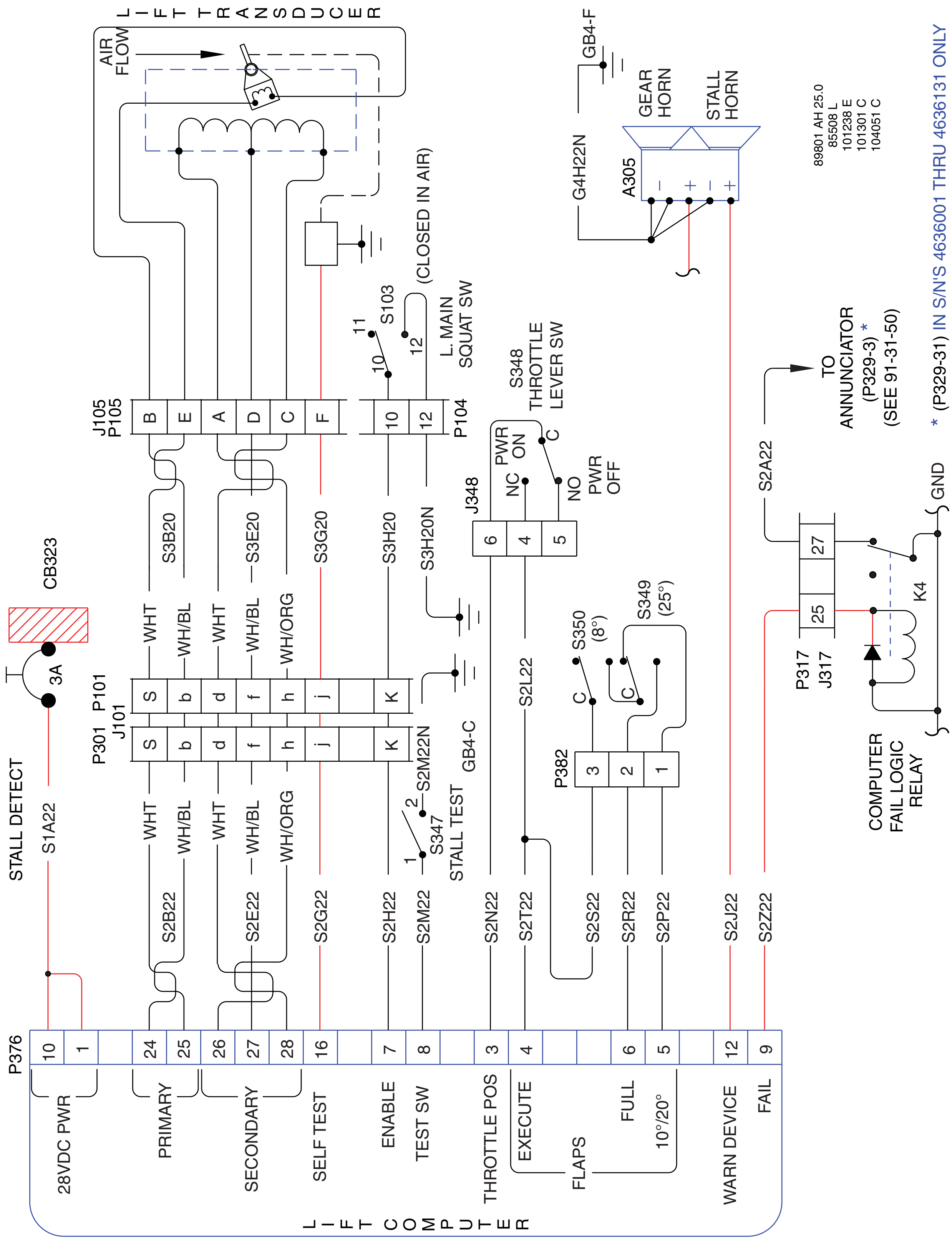
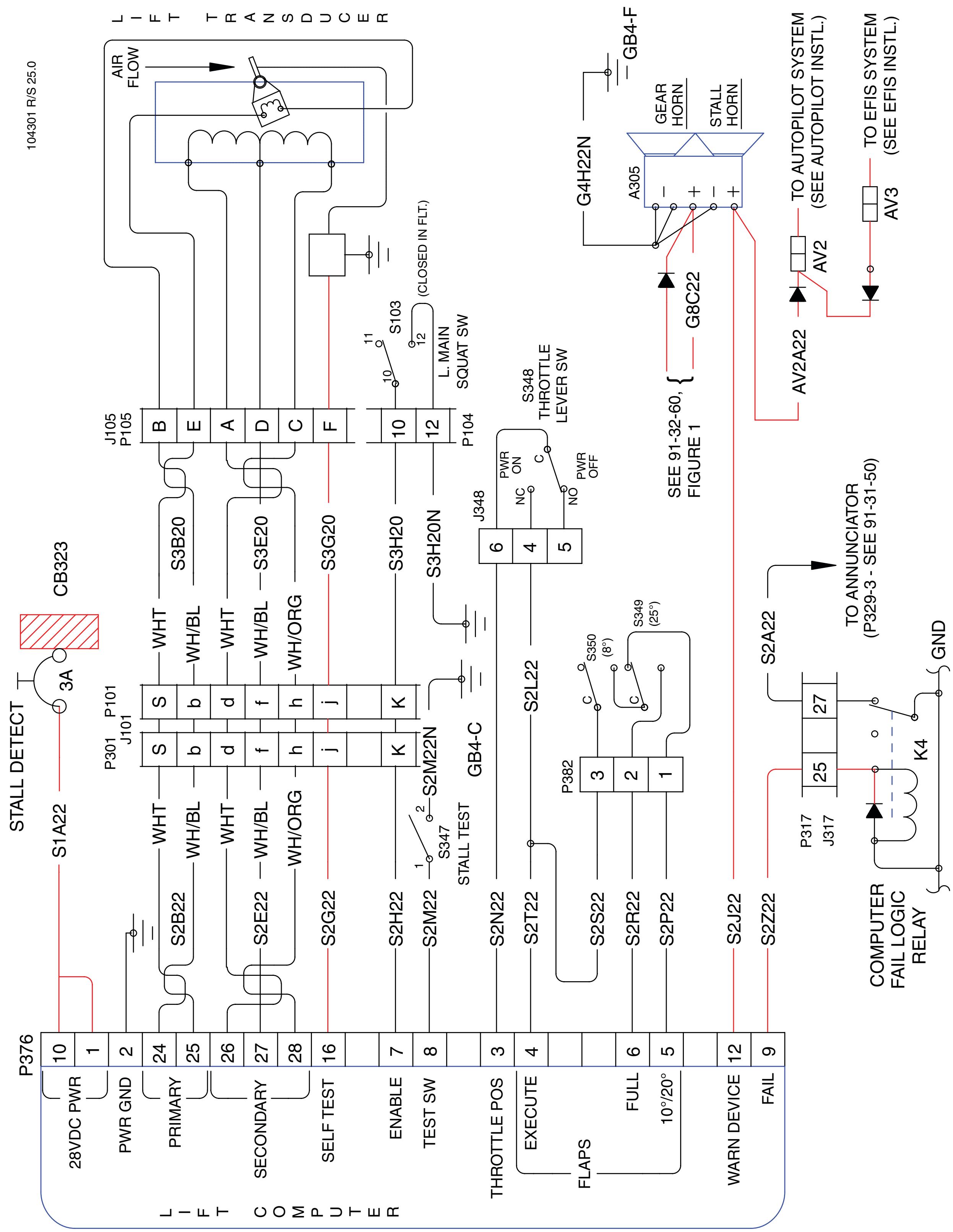
Item #	Designation	Description
1	CB323, CB28*	Circuit Breaker - Stall Detect/Warn (3 Amp)
2	S347	Stall Test Switch
3	S103	Squat Switch Assembly
4	S349, S350	Actuator Switch (2 each)
5	S348	Throttle Lever Switch Assembly
6	A305	Horn - Dual Warn Unit
7	K4	Relay Assembly
8		Lift Computer
9		Lift Transducer



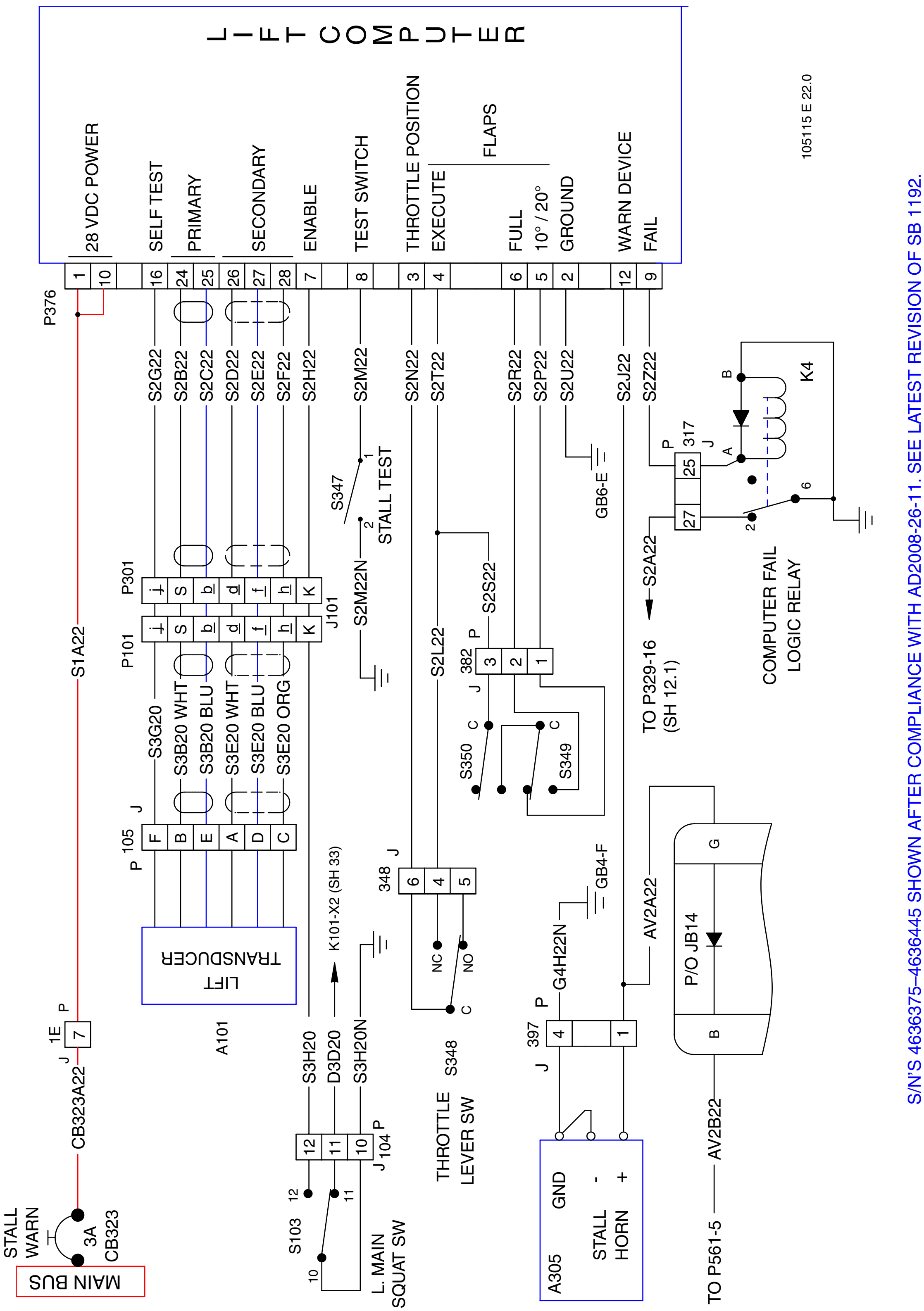
Stall Detection and Warning
Figure 1 (Sheet 1 of 8)



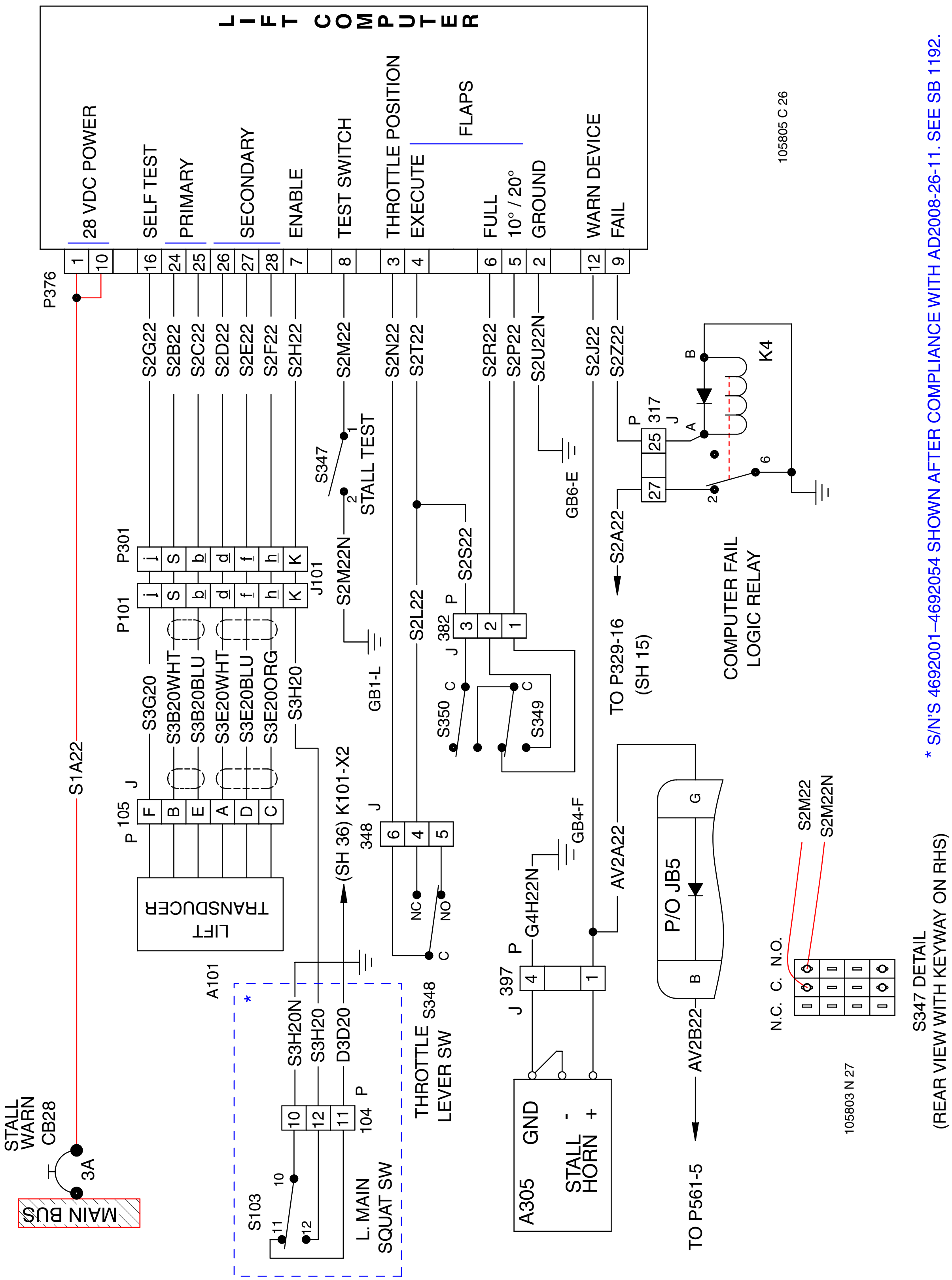
Stall Detection and Warning
Figure 1 (Sheet 2 of 8)



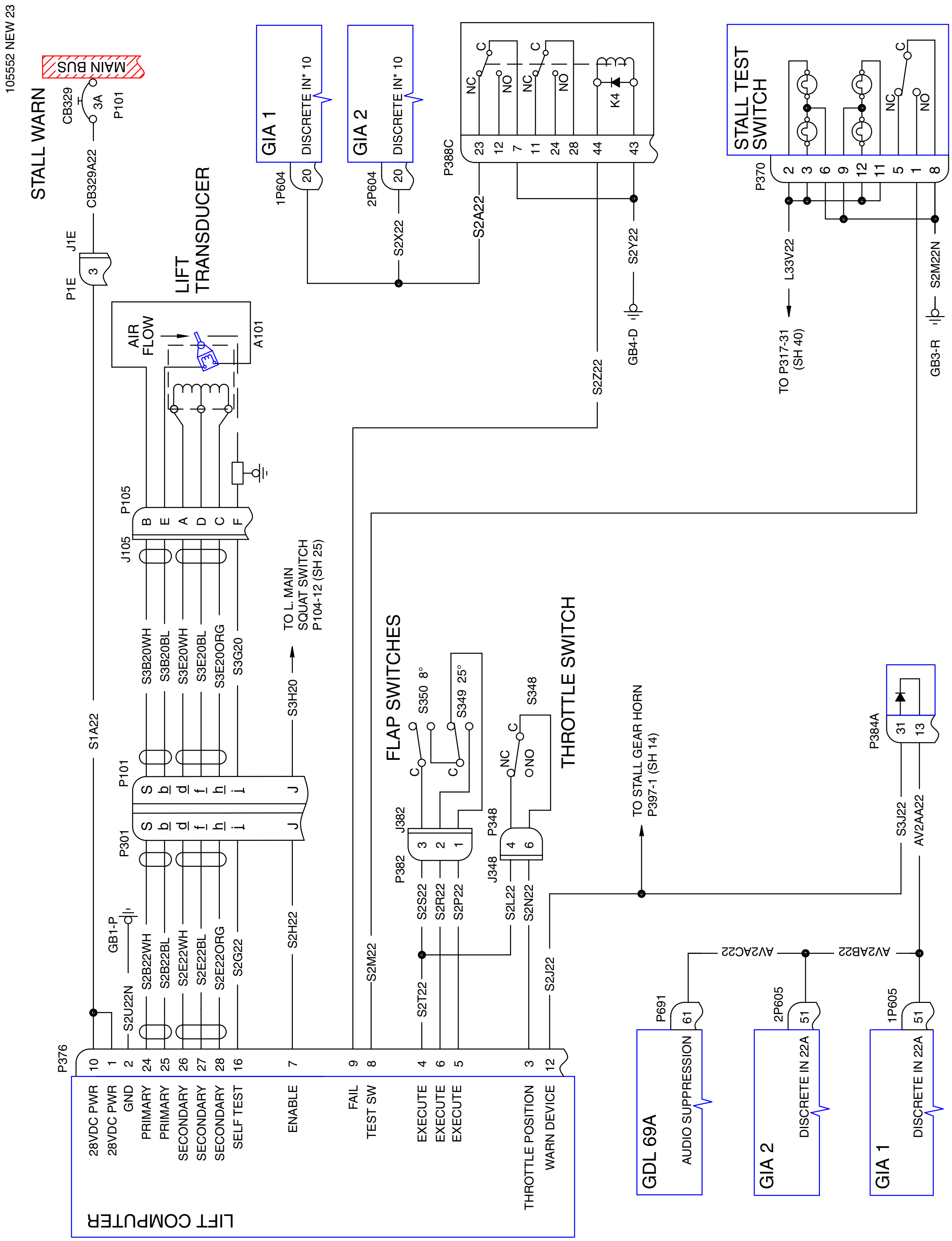
Stall Detection and Warning
Figure 1 (Sheet 3 of 8)



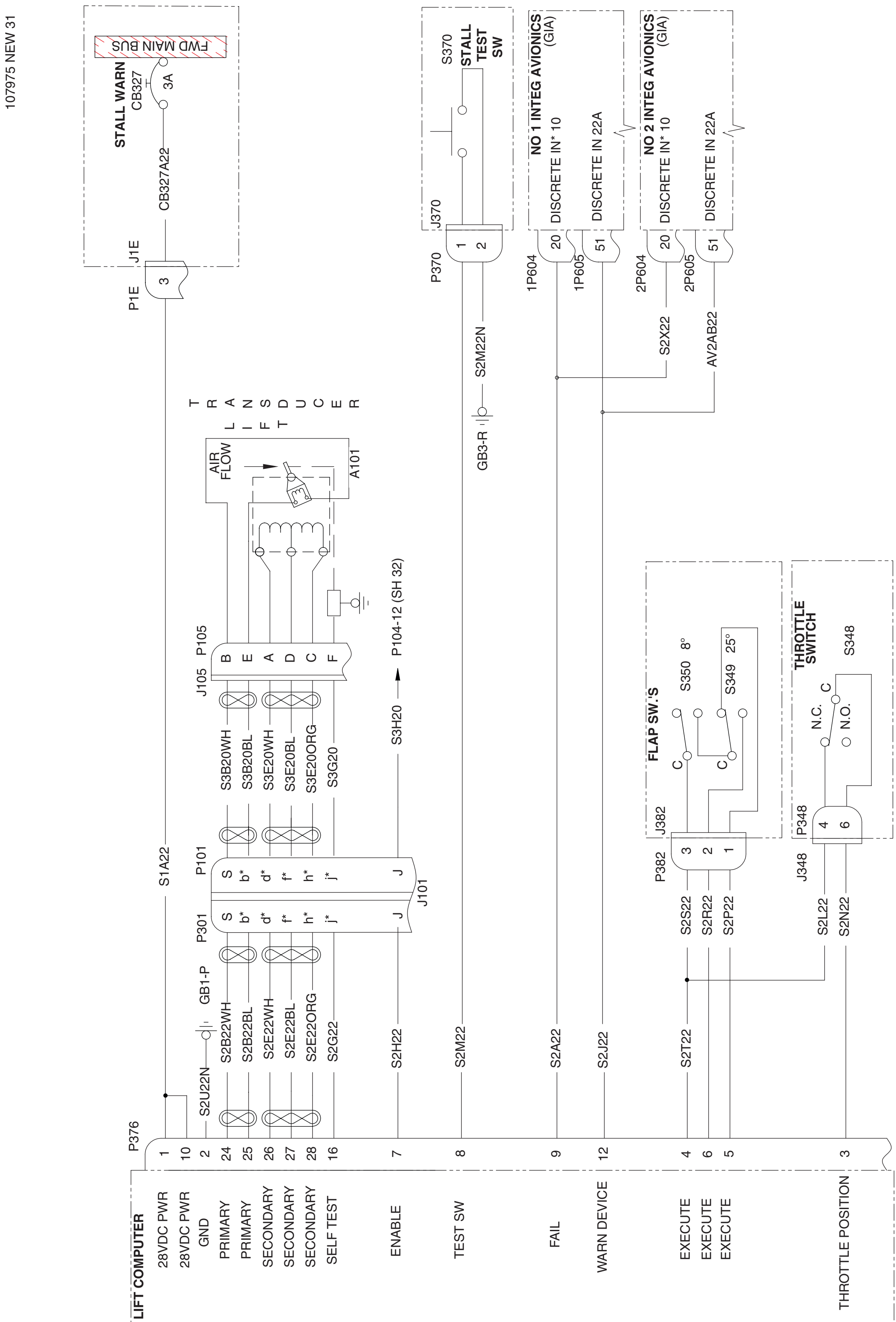
Stall Detection and Warning
Figure 1 (Sheet 4 of 8)



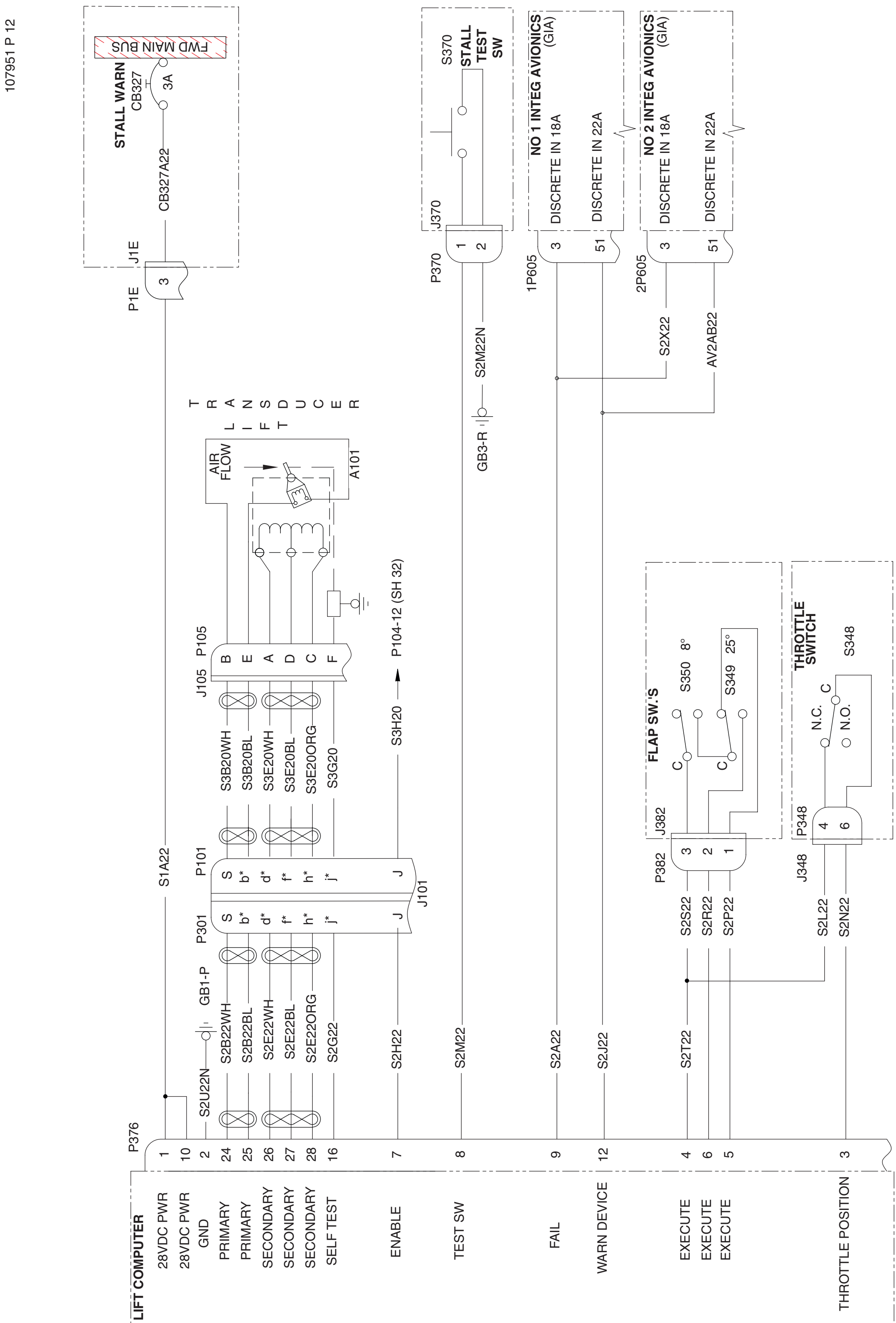
Stall Detection and Warning
Figure 1 (Sheet 5 of 8)



Stall Detection and Warning
Figure 1 (Sheet 6 of 8)

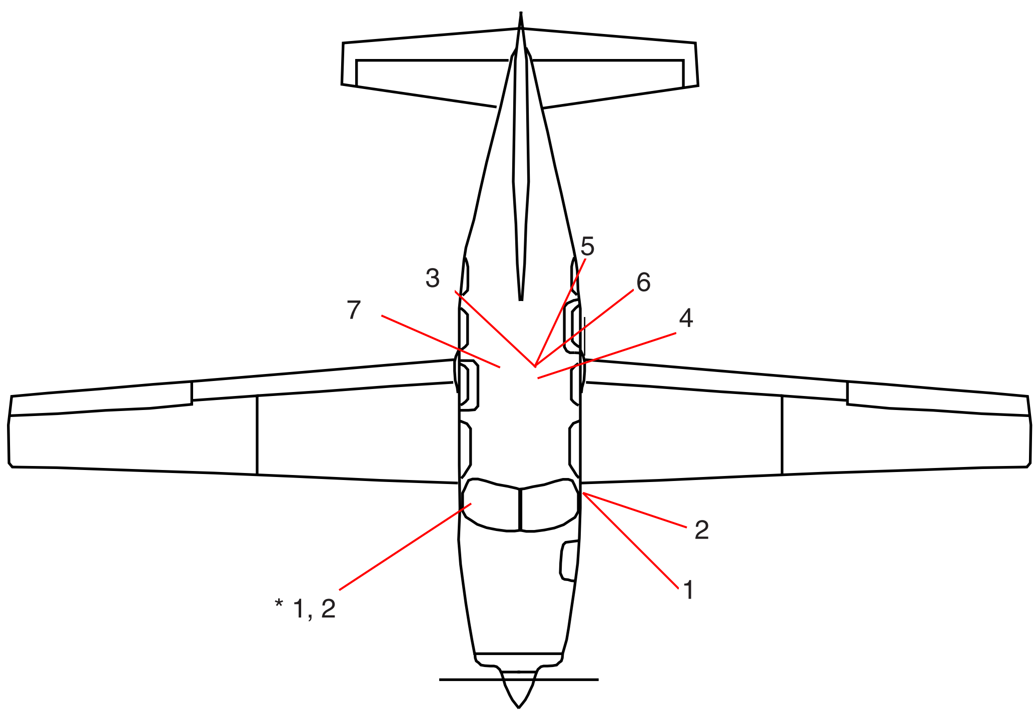


Stall Detection and Warning
Figure 1 (Sheet 7 of 8)

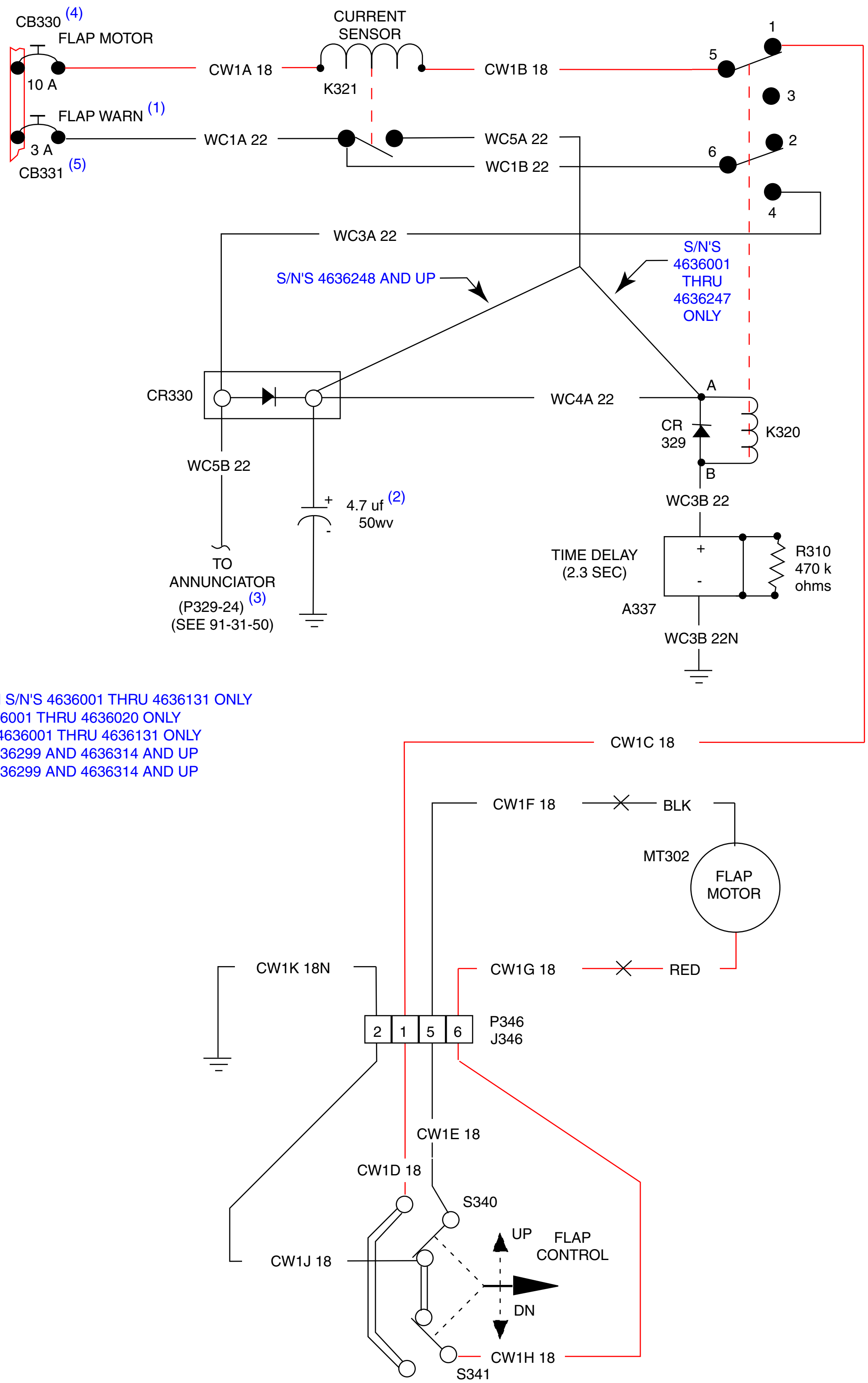


Stall Detection and Warning
Figure 1 (Sheet 7 of 8)

Item #	Designation	Description
1	CB330, CB329, CB53*	Circuit Breaker - Flap Motor (10 Amp)
2	CB331, CB330, CB54*	Circuit Breaker - Flap Warn (3 Amp)
3	K321	Current Sensor
4	MT302	Actuator Motor Assembly
5	K320	Relay
6	A337	Time Delay Module
7	S340, S341	Flap Control Switch
	CR330	Diode Assembly
	R310	Resistor Assembly
		Capacitor Assembly



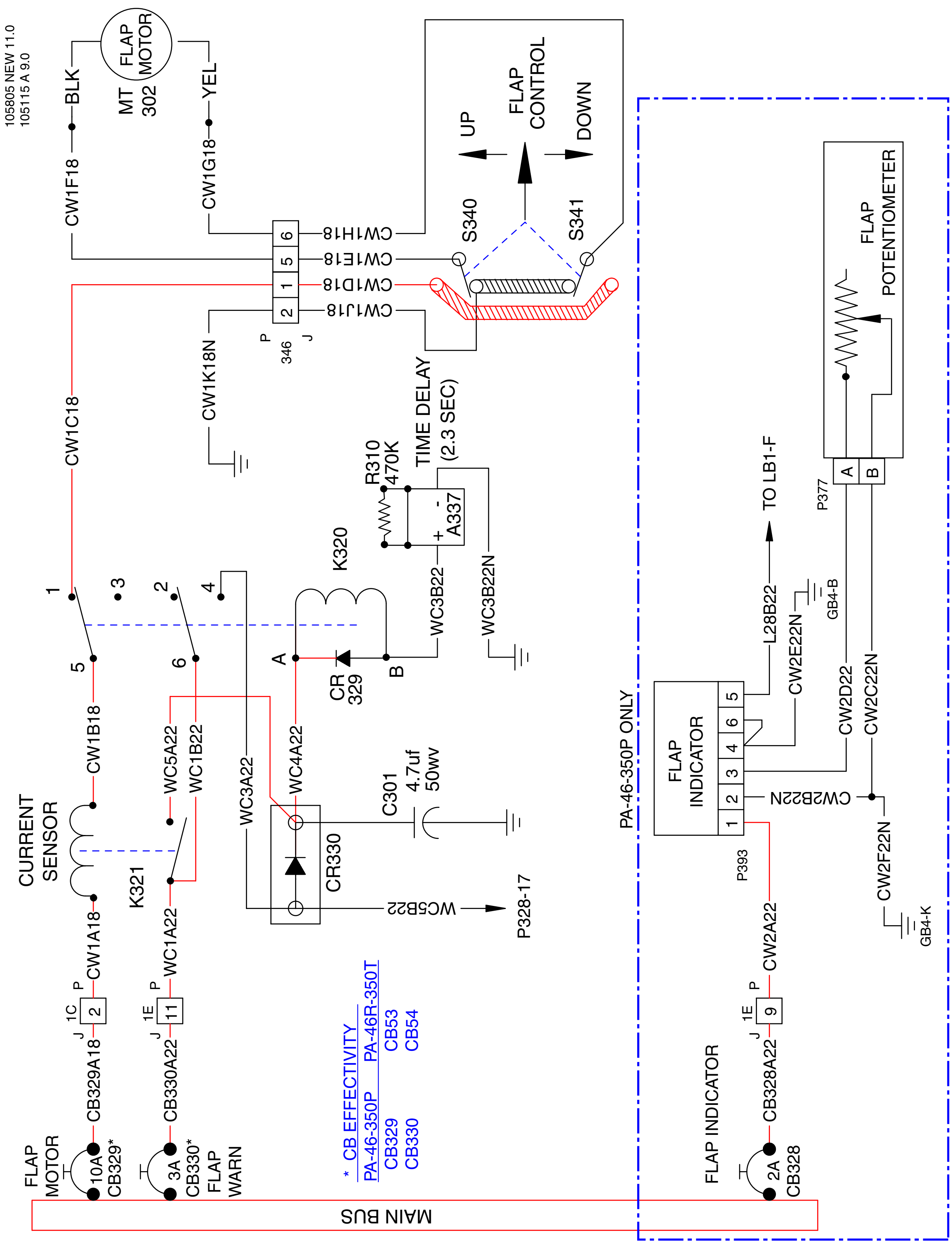
Electric Flaps
Figure 1 (Sheet 1 of 6)



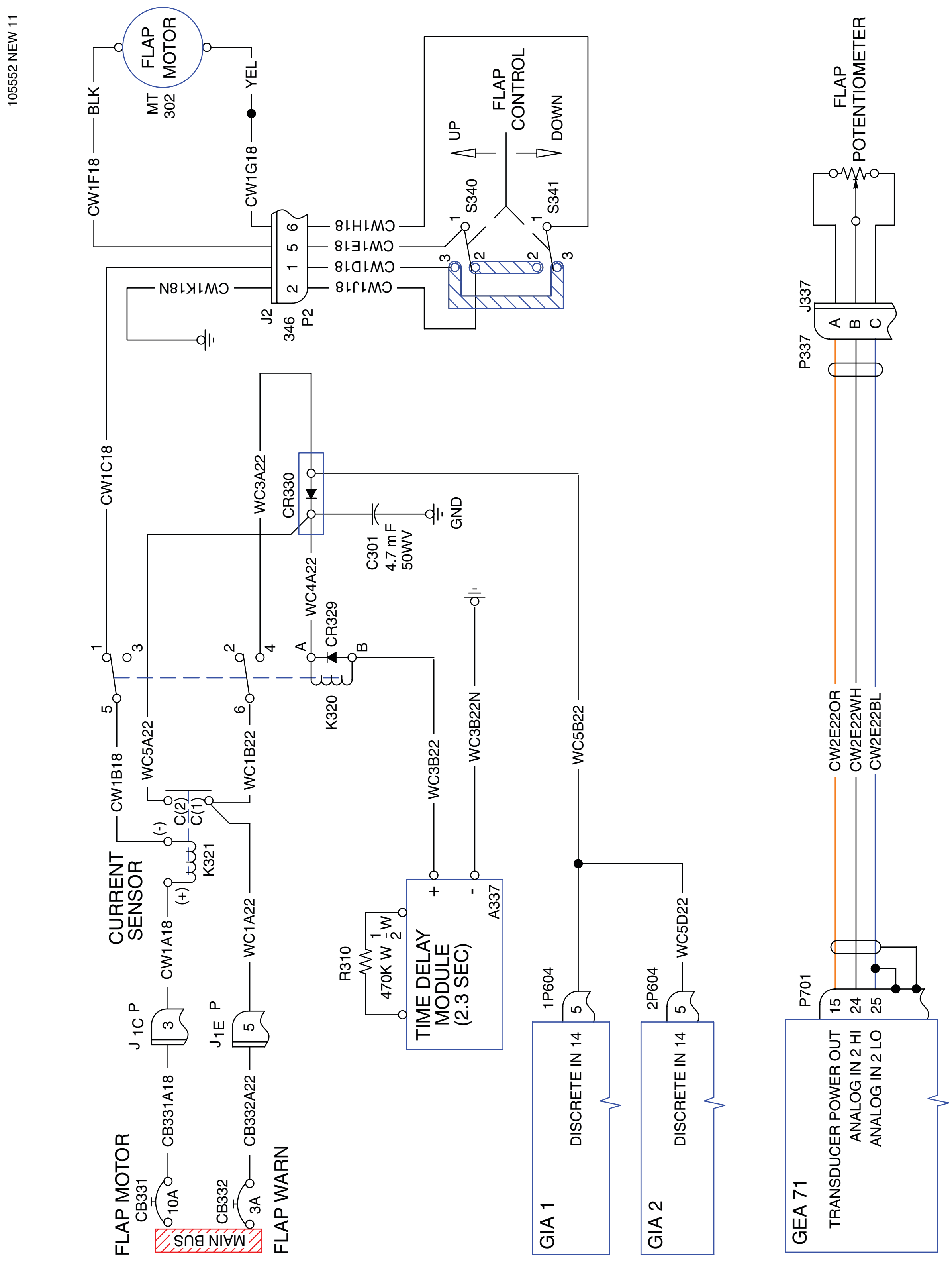
(1) - "FLAP ANNUN." IN S/N'S 4636001 THRU 4636131 ONLY
(2) - 5.0 uf IN S/N'S 4636001 THRU 4636020 ONLY
(3) - (P328-9) IN S/N'S 4636001 THRU 4636131 ONLY
(4) - CB329 IN S/N'S 4636299 AND 4636314 AND UP
(5) - CB330 IN S/N'S 4636299 AND 4636314 AND UP

89801 AH 11.0
85508 L
101238 E
101301 C
104051 C
104301 NEW/S

Electric Flaps
Figure 1 (Sheet 2 of 6)

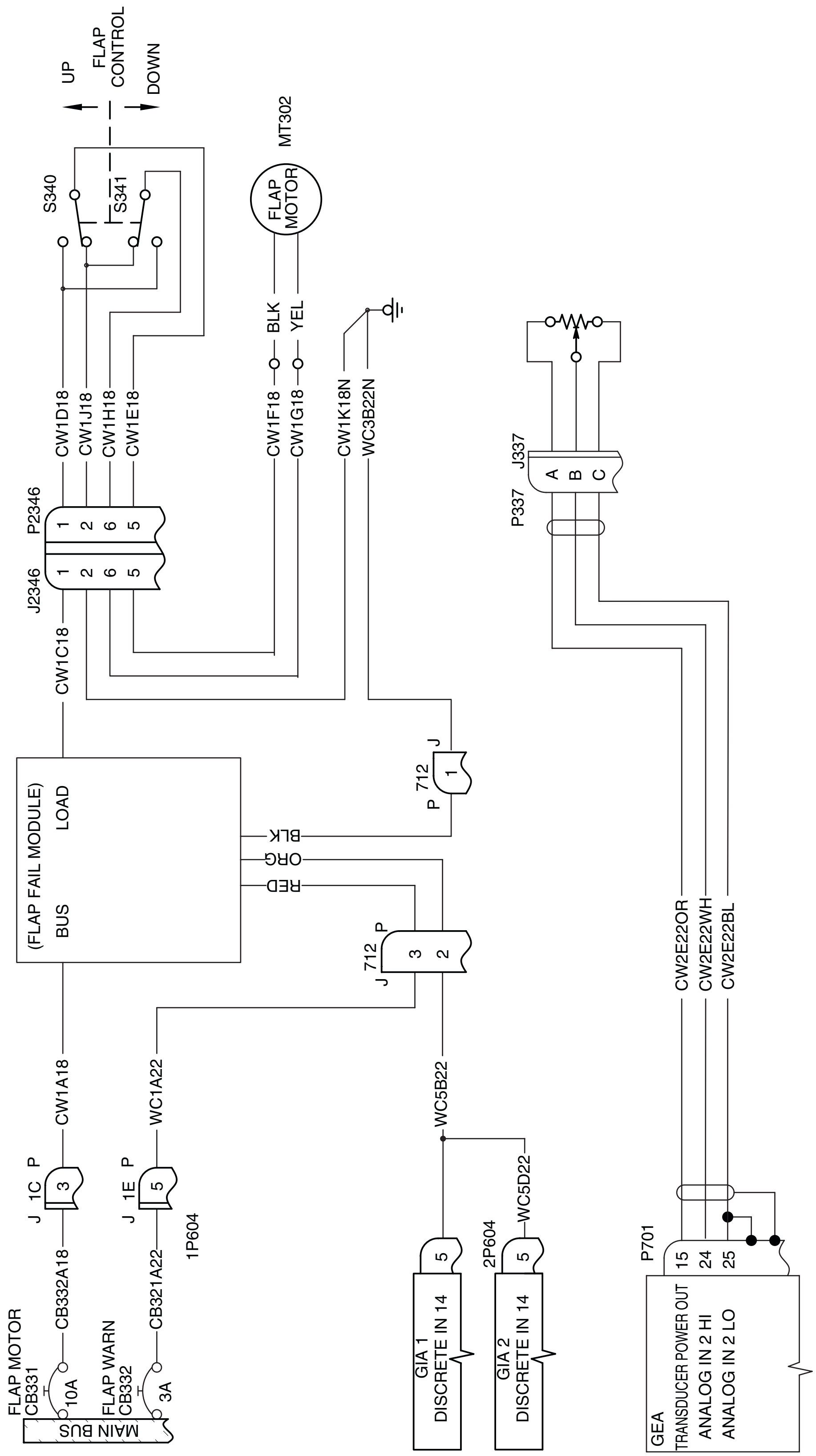


Electric Flaps
Figure 1 (Sheet 3 of 6)

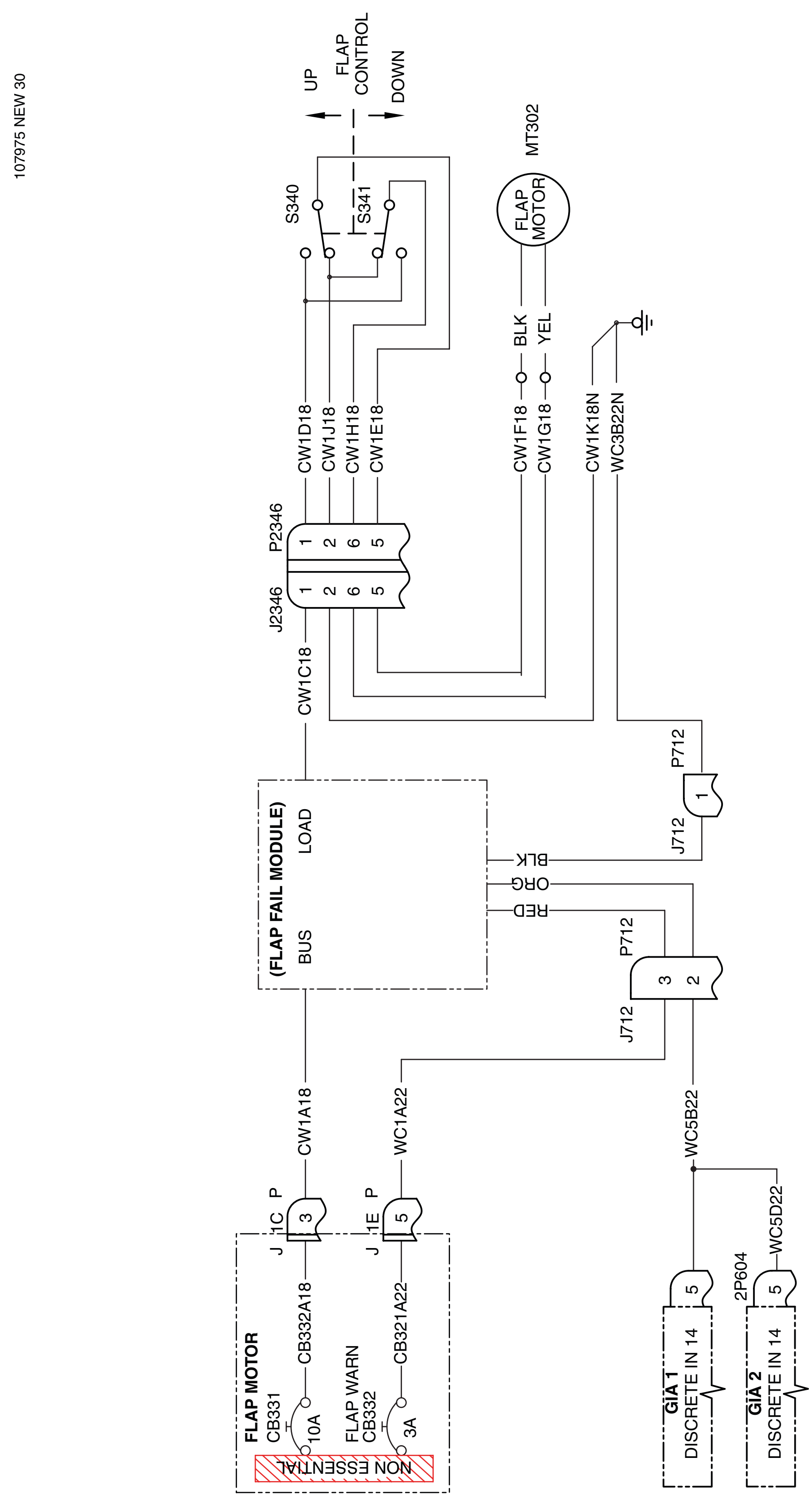


Electric Flaps
Figure 1 (Sheet 4 of 6)

105552 D 11

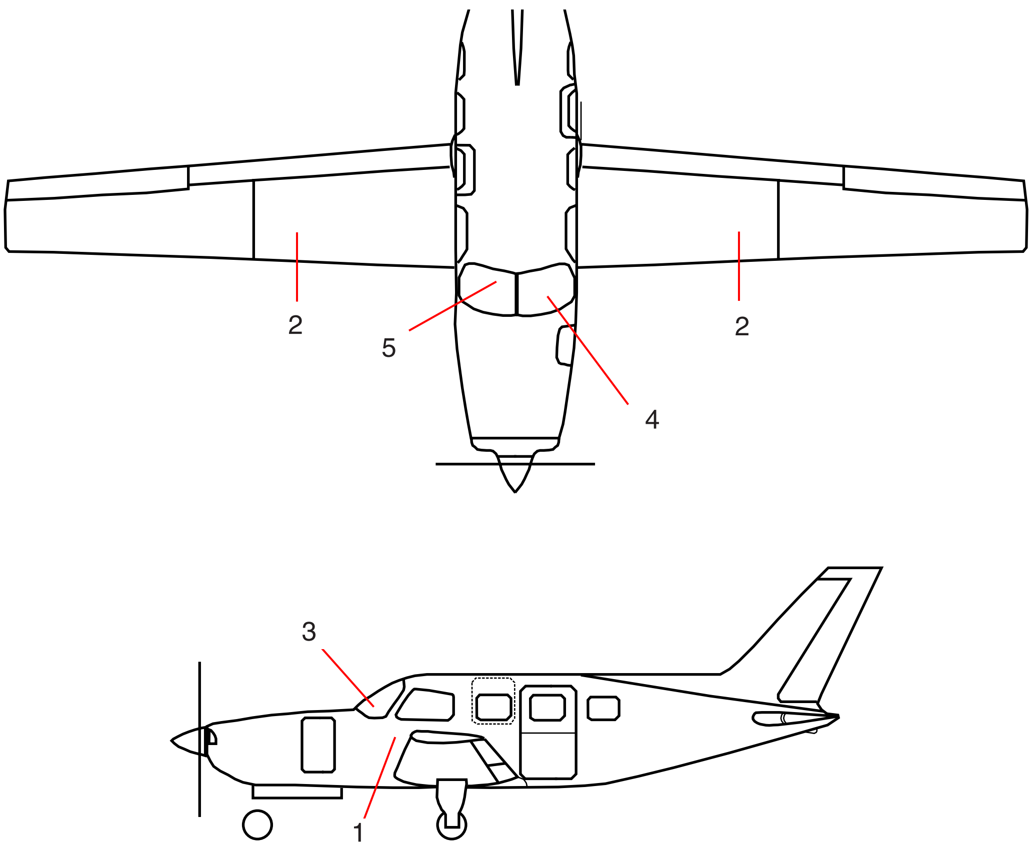


Electric Flaps
Figure 1 (Sheet 5 of 6)

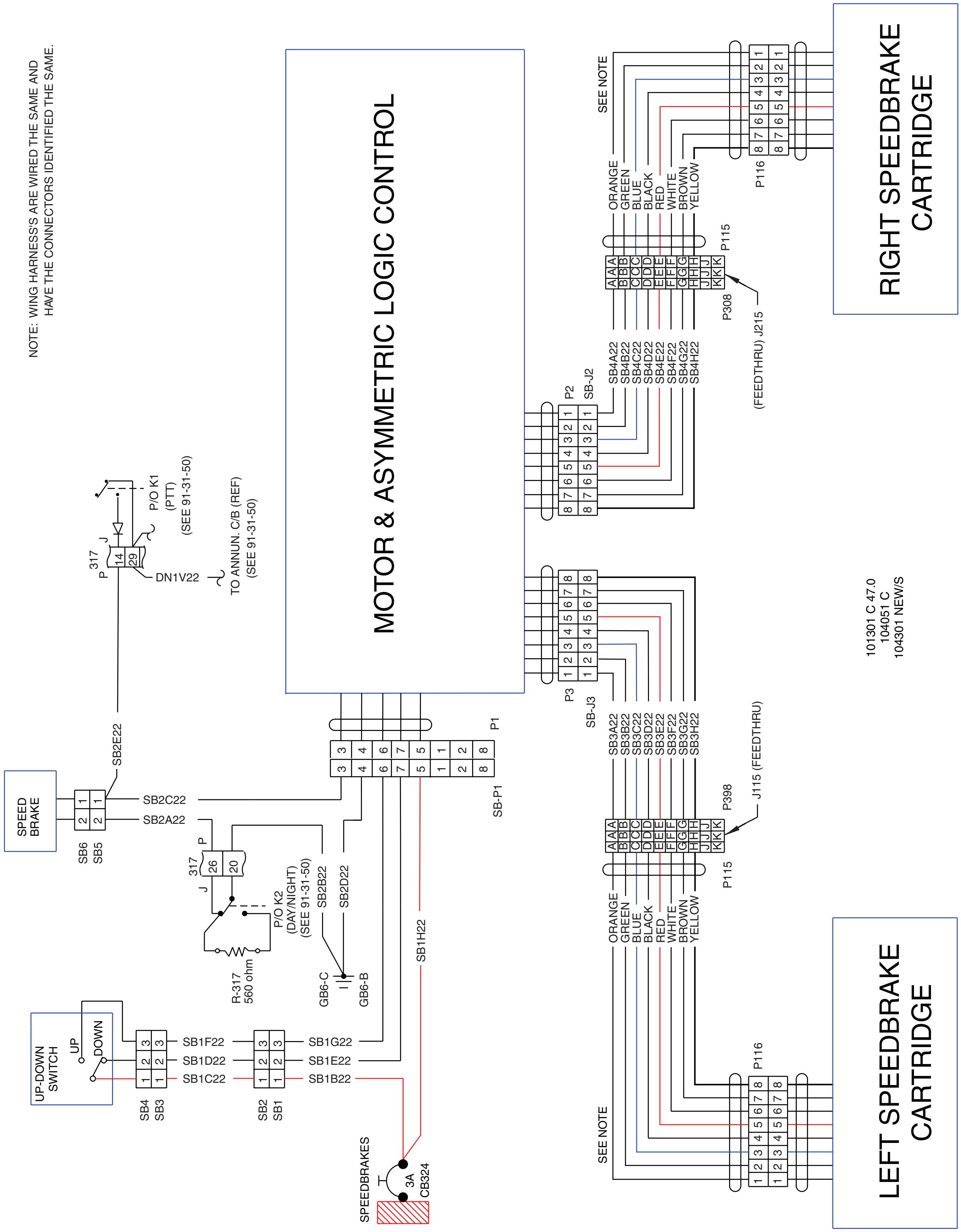


Electric Flaps
Figure 1 (Sheet 6 of 6)

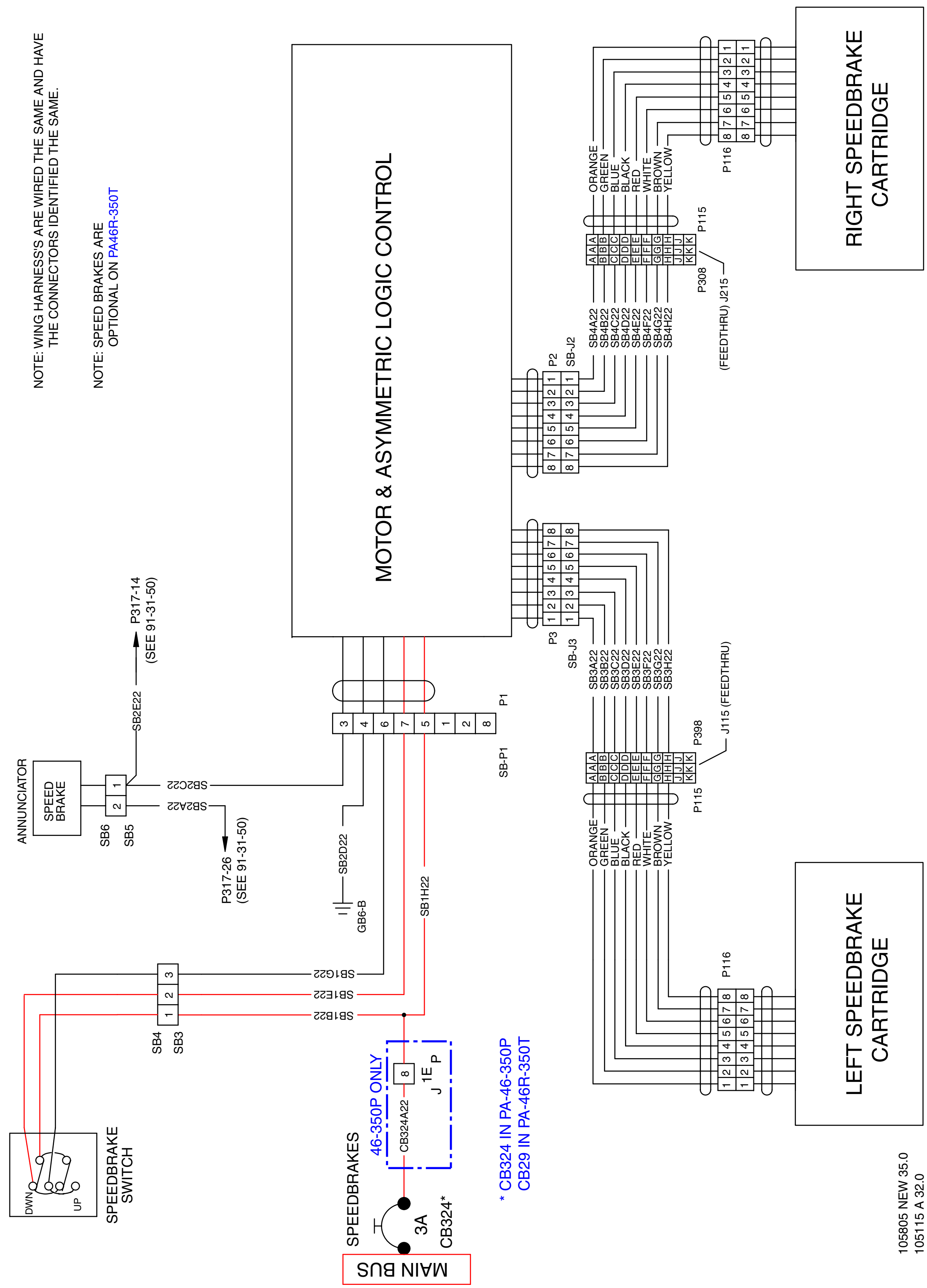
Item #	Designation	Description
1	CB324	Circuit Breaker - Speedbrakes (3 Amp)
2		SpeedBrake Cartridge(s)
3		Up/Down Switch (on yoke)
4		Annunciator Light
5		Asymmetric Logic Switching Unit



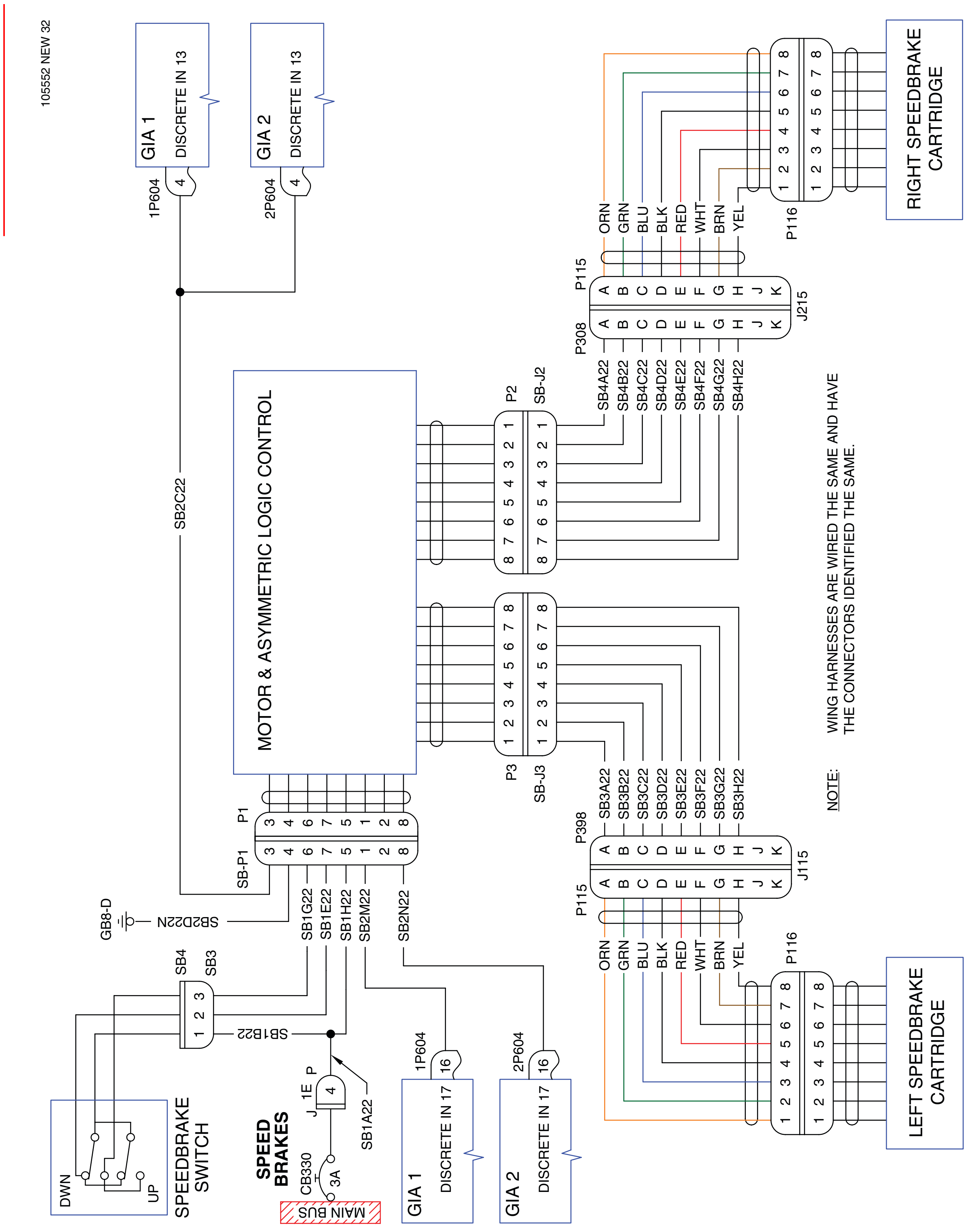
SpeedBrakes
Figure 1 (Sheet 1 of 5)



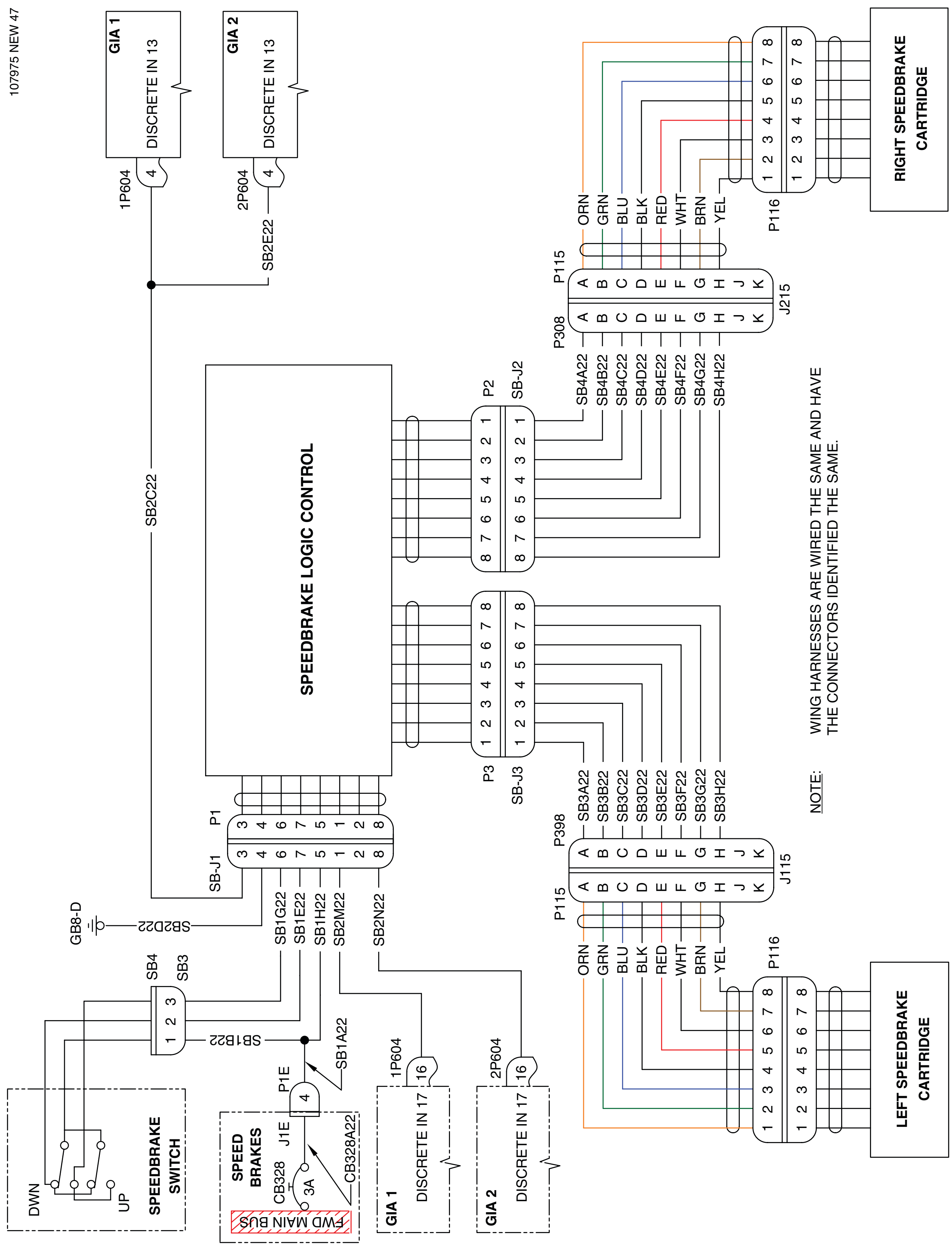
SpeedBrakes
Figure 1 (Sheet 2 of 5)



SpeedBrakes
Figure 1 (Sheet 3 of 5)



SpeedBrakes
Figure 1 (Sheet 4 of 5)



SpeedBrakes
Figure 1 (Sheet 5 of 5)

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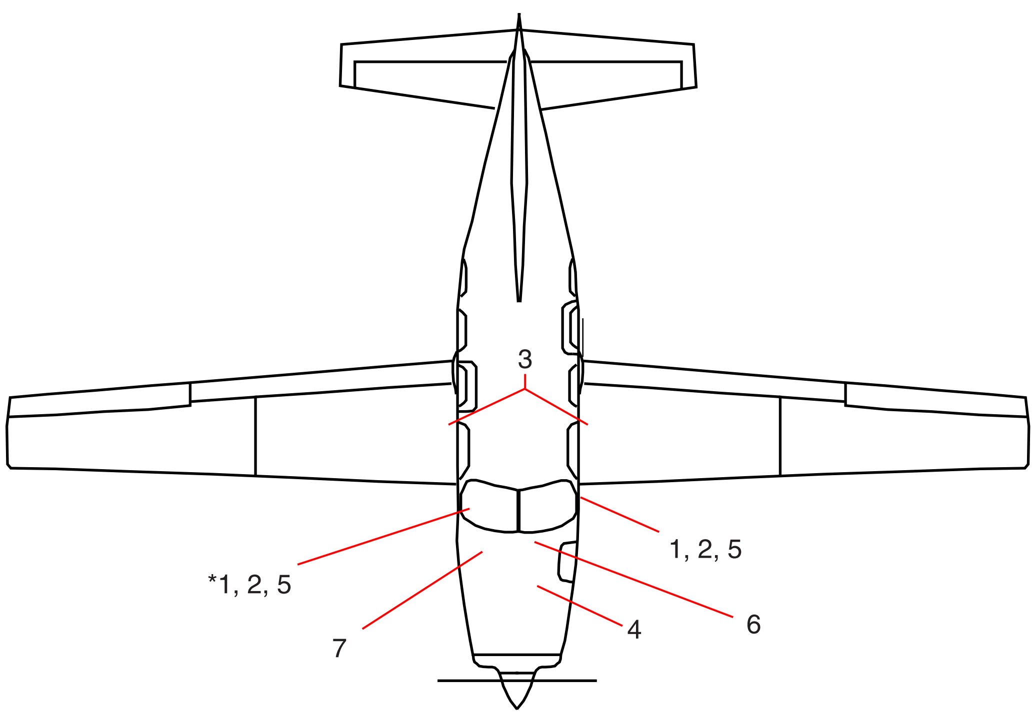
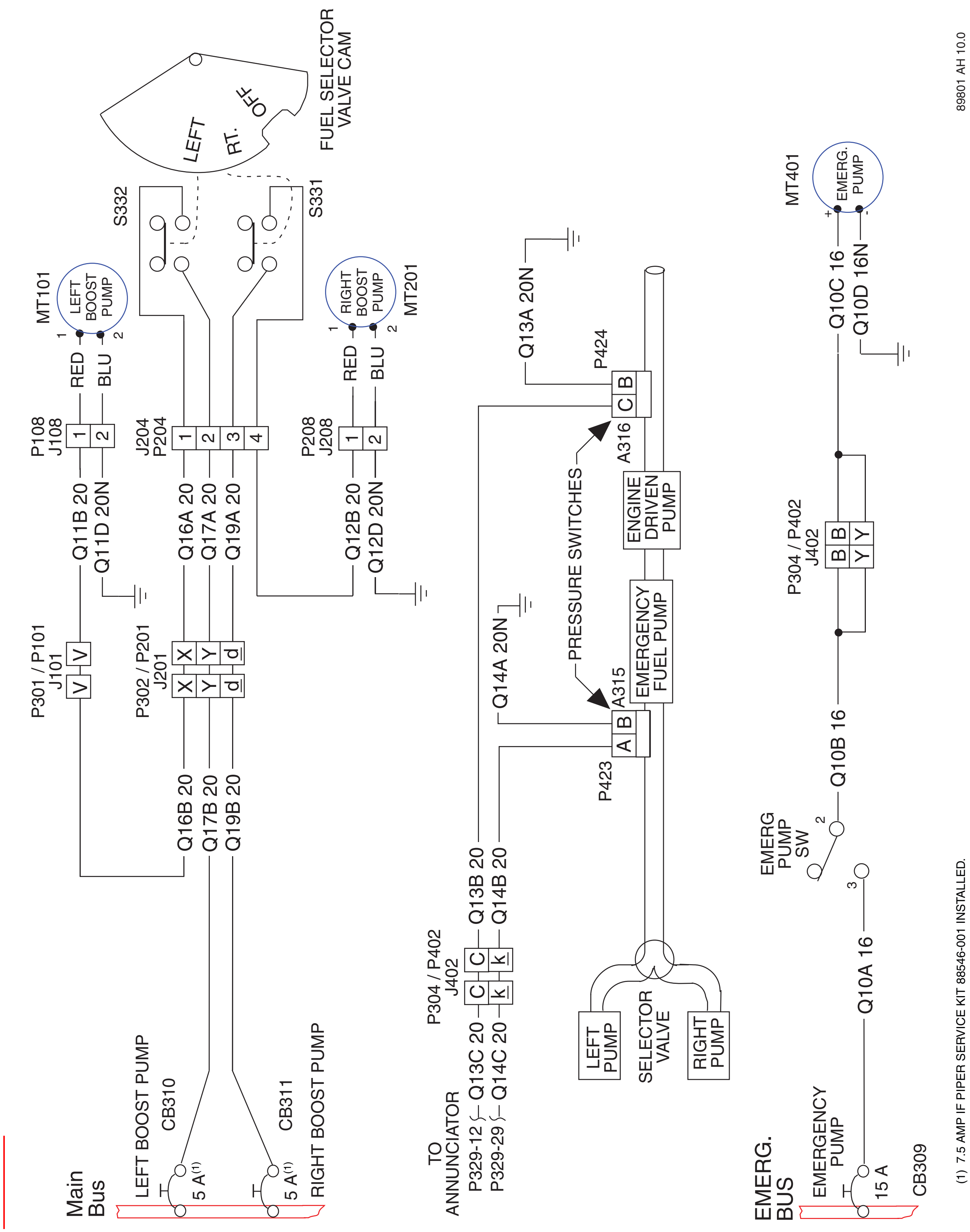
Item #	Designation	Description
1	CB310, CB12(1)	Circuit Breaker - L Boost Pump (5 Amp)
2	CB311, CB13(1)	Circuit Breaker - R Boost Pump (5 Amp)
3	MT101, MT201	Fuel Pump
4	A315, A316	Fuel Pressure Switch
5	CB309, CB11, CB317	Circuit Breaker - Emergency Fuel Pump (15 Amp)
6	S305	Emergency Fuel Pump Switch
7	MT401	Emergency Fuel Pump

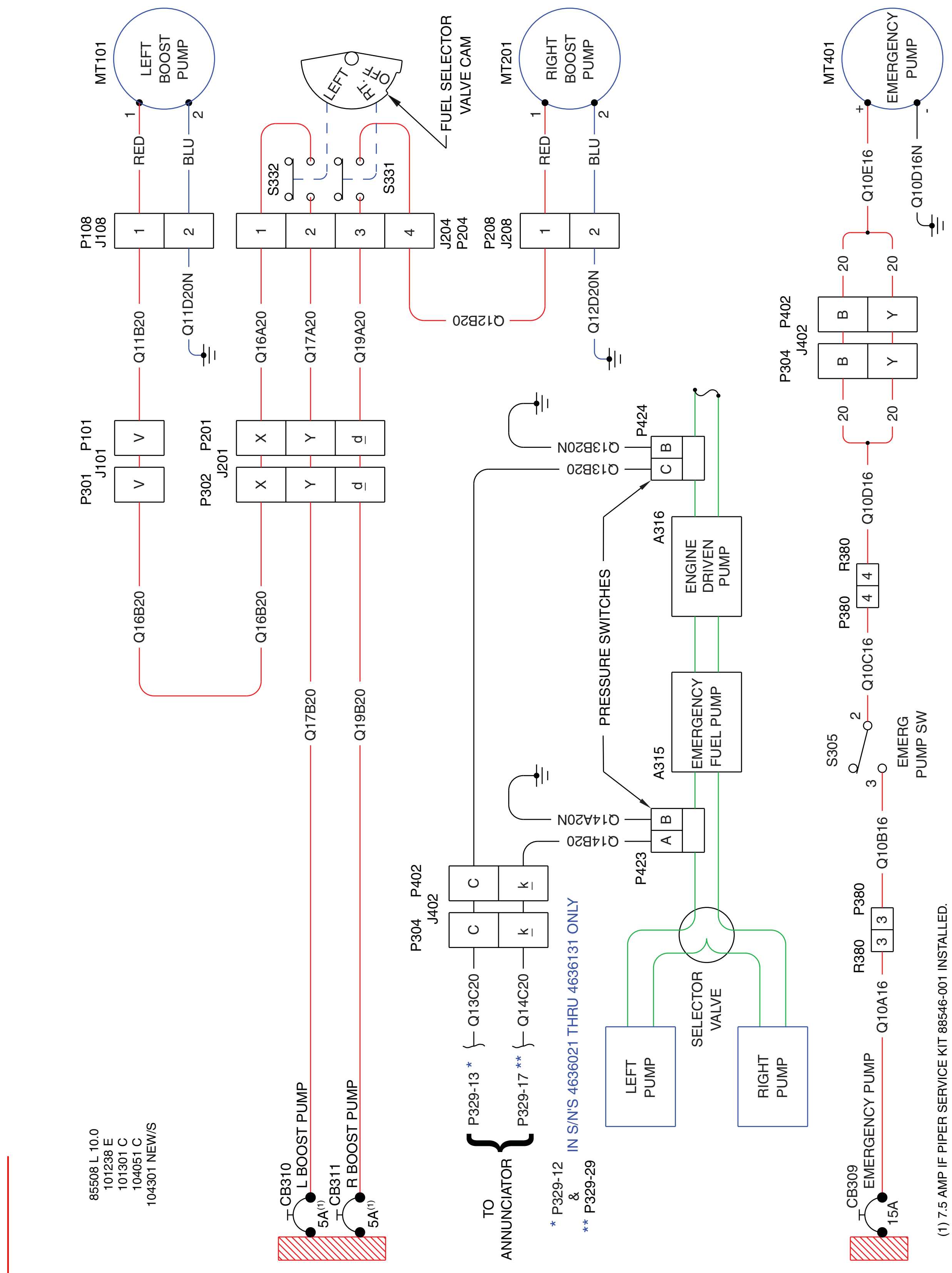
(1) 7.5 amp in S/N's 4636658 and up / 4692214 and up; or if Piper Service Kit 88546-001 installed in earlier airplanes.



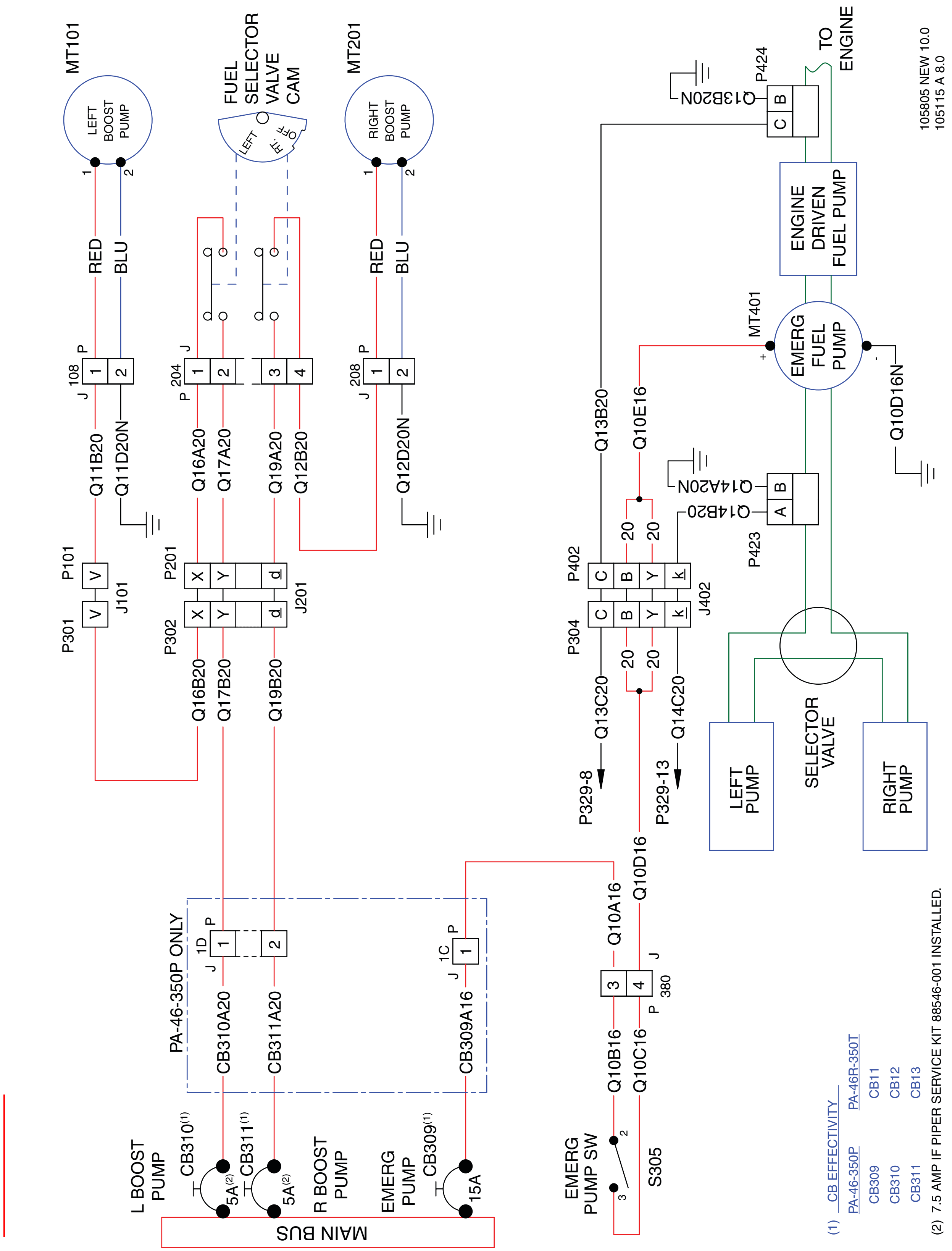
Fuel Pumps
Figure 1 (Sheet 1 of 5)



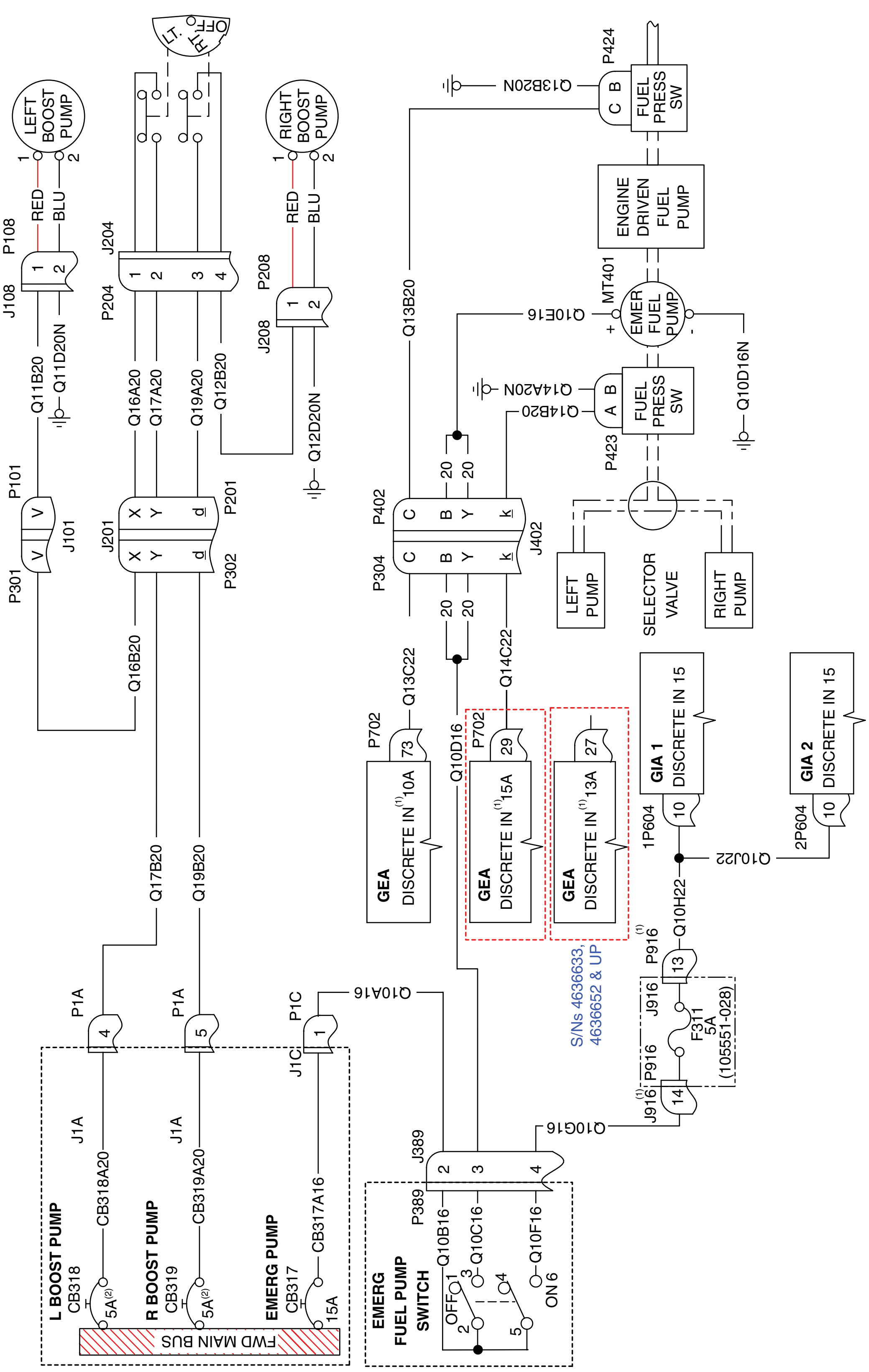
Fuel Pumps
Figure 1 (Sheet 2 of 5)



Fuel Pumps
Figure 1 (Sheet 3 of 5)



Fuel Pumps
Figure 1 (Sheet 4 of 5)



(1) P/J916 - CONNECTOR IN S/N's 4636505 & UP AND 4692167 & UP. SEE ALSO 107975 SH 15.
(2) 7.5 AMP IN S/N's 4636747 & UP AND 4692214 & UP. AUTHORIZED SERVICE REPLACEMENT IN S/N's 4636658–4636746; OR IF PIPER SERVICE KIT 88546-001 INSTALLED IN S/N's 4636001–4636657 OR 4692001–4692213.

105552 C 9
107975 NEW 15
107956 E

Fuel Pumps
Figure 1 (Sheet 5 of 5)

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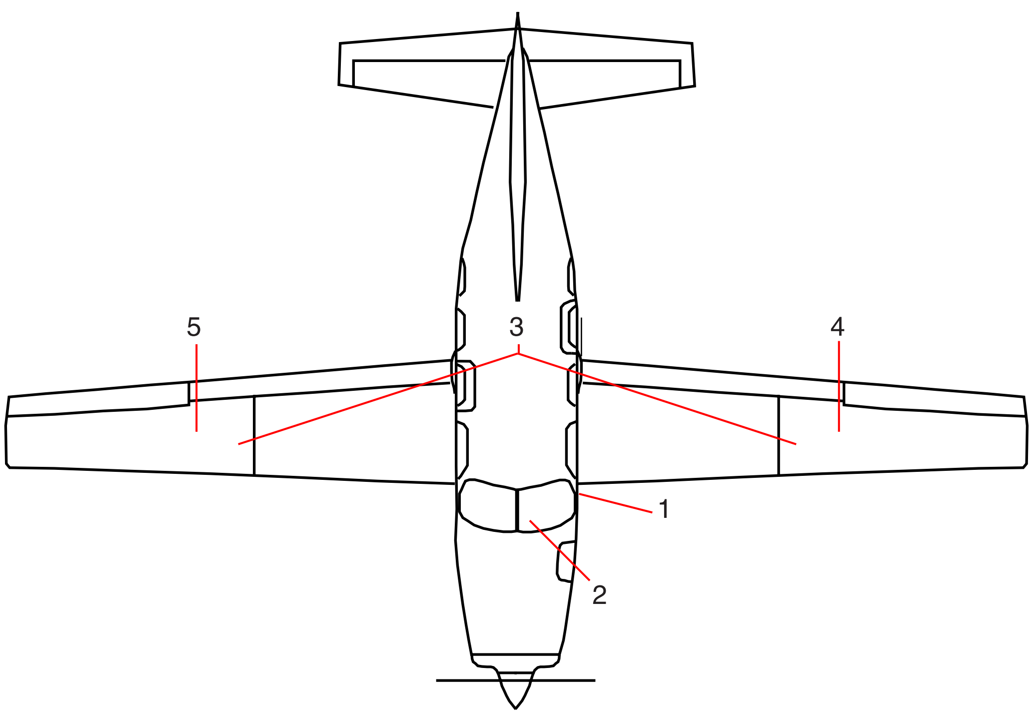
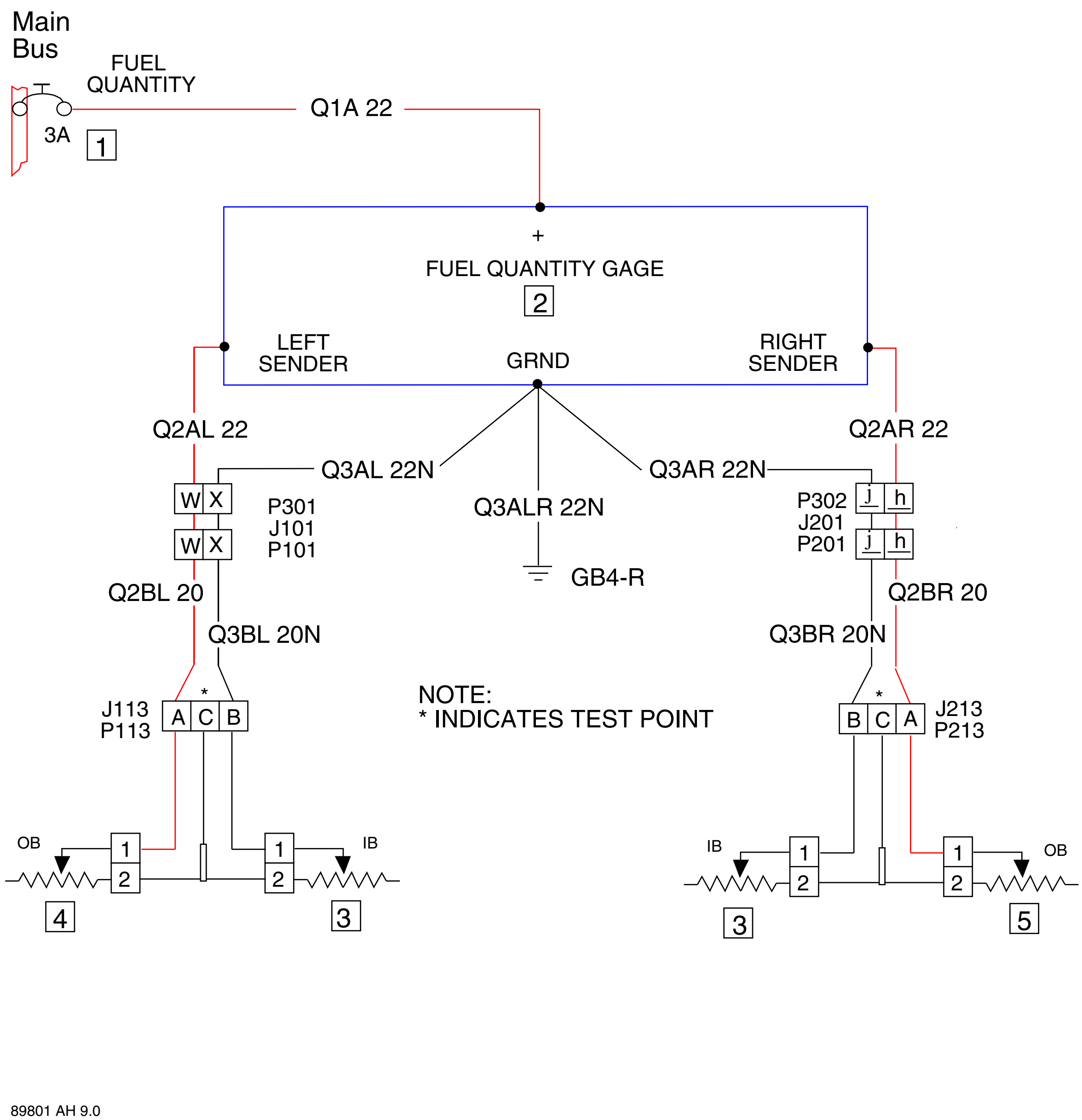
Item #	Designation	Description
1	CB312 *	Circuit Breaker (3 Amp)
	or	
	CB372	Circuit Breaker (5 Amp)
2	M305	Dual Fuel Indicator
3		Inboard Fuel Quantity Senders
4		Outboard Fuel Quantity Sender, Left
5		Outboard Fuel Quantity Sender, Right

* S/N's 4636001 thru 4636020 only

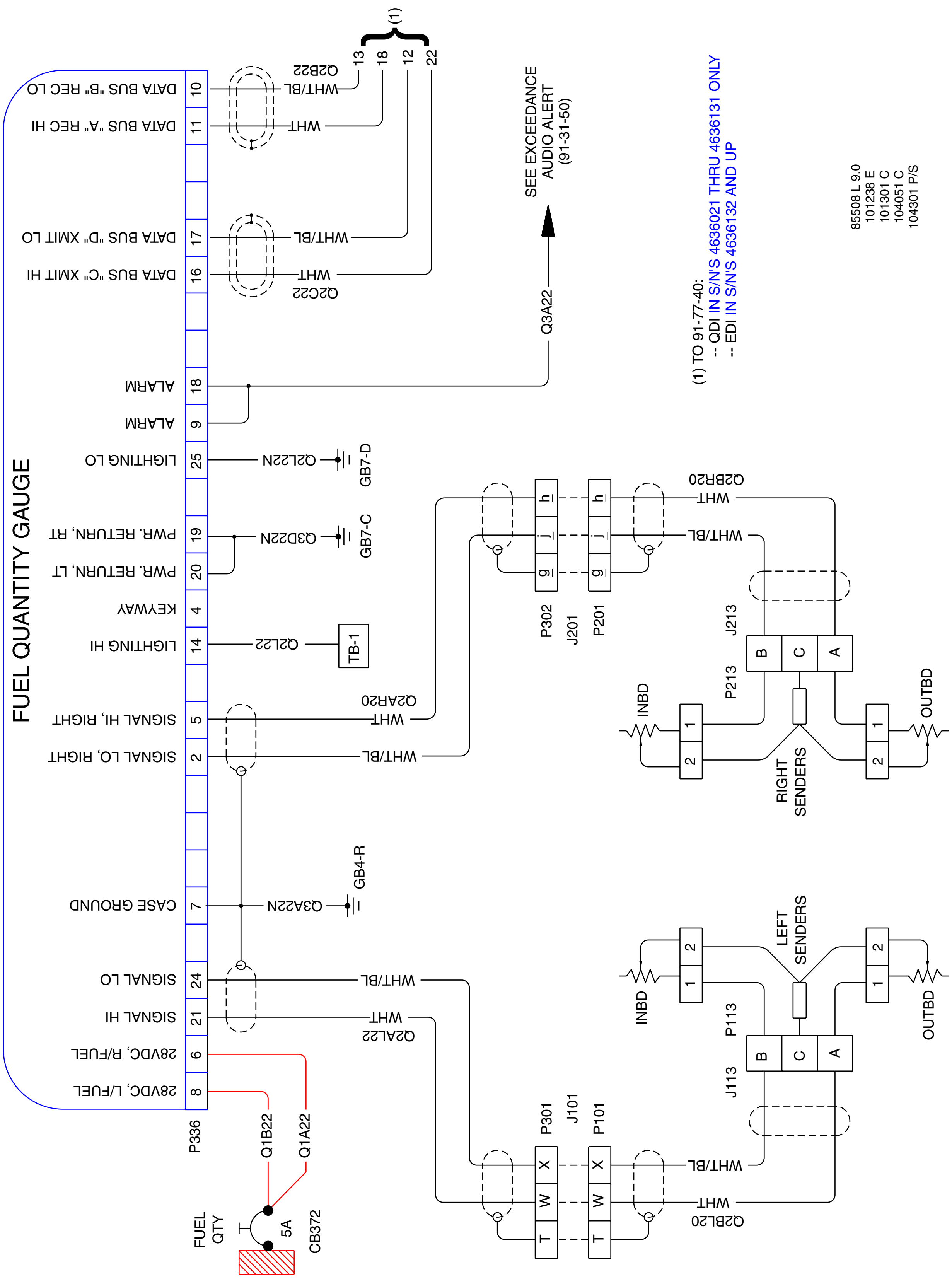
In S/N's 4636375–4636459, 4636461–4636462, 4636481, see 105115 Sheet 5.
In S/N's 4692001–4692133, 4692141, 4692149, 4692153, see 105805 Sheet 7.



Fuel Quantity Indicator
Figure 1 (Sheet 1 of 7)

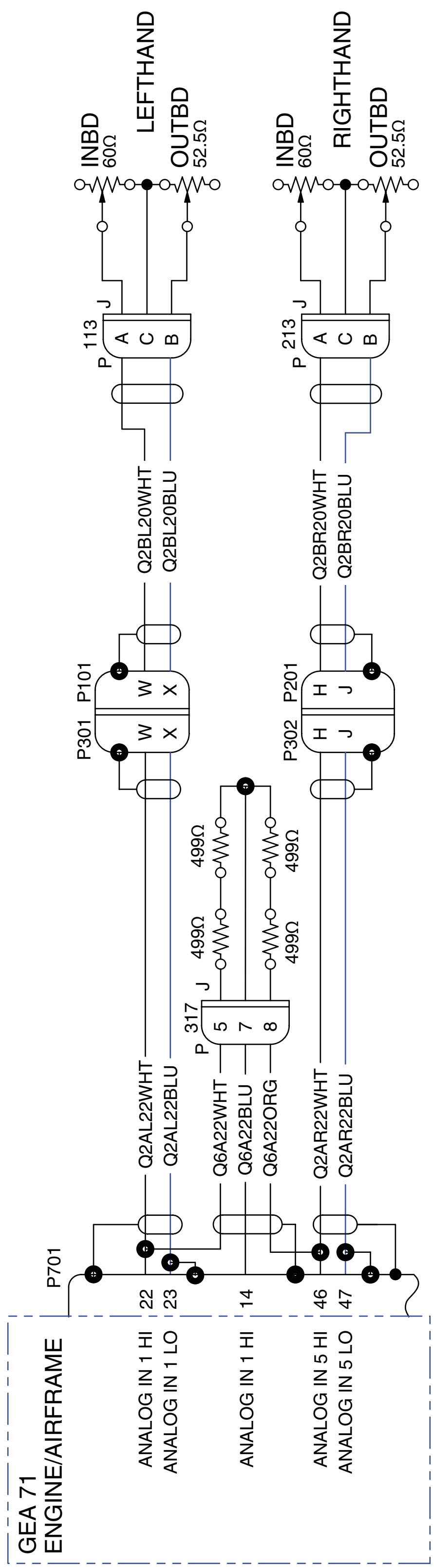


Fuel Quantity Indicator
Figure 1 (Sheet 2 of 7)

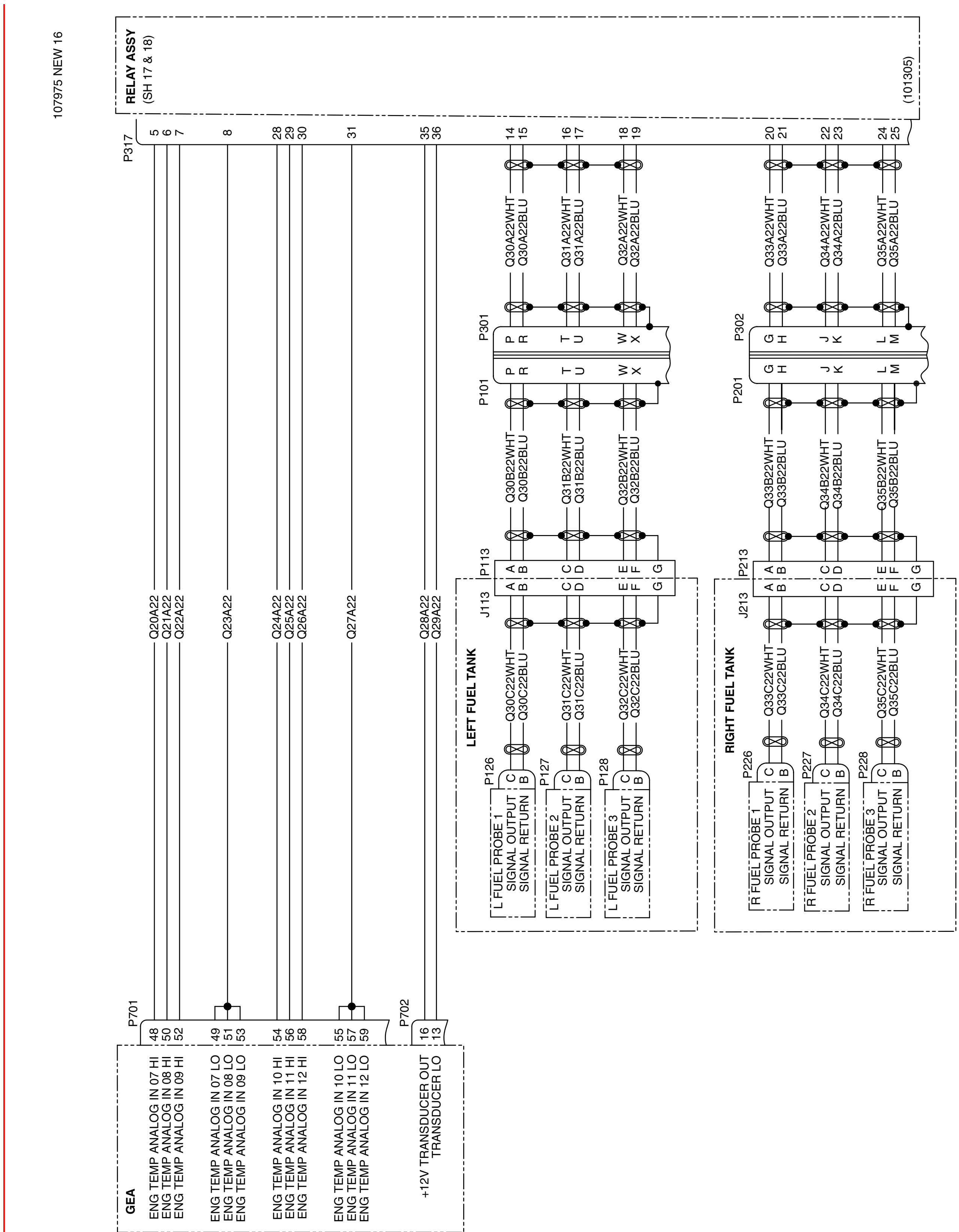


Fuel Quantity Indicator
Figure 1 (Sheet 3 of 7)

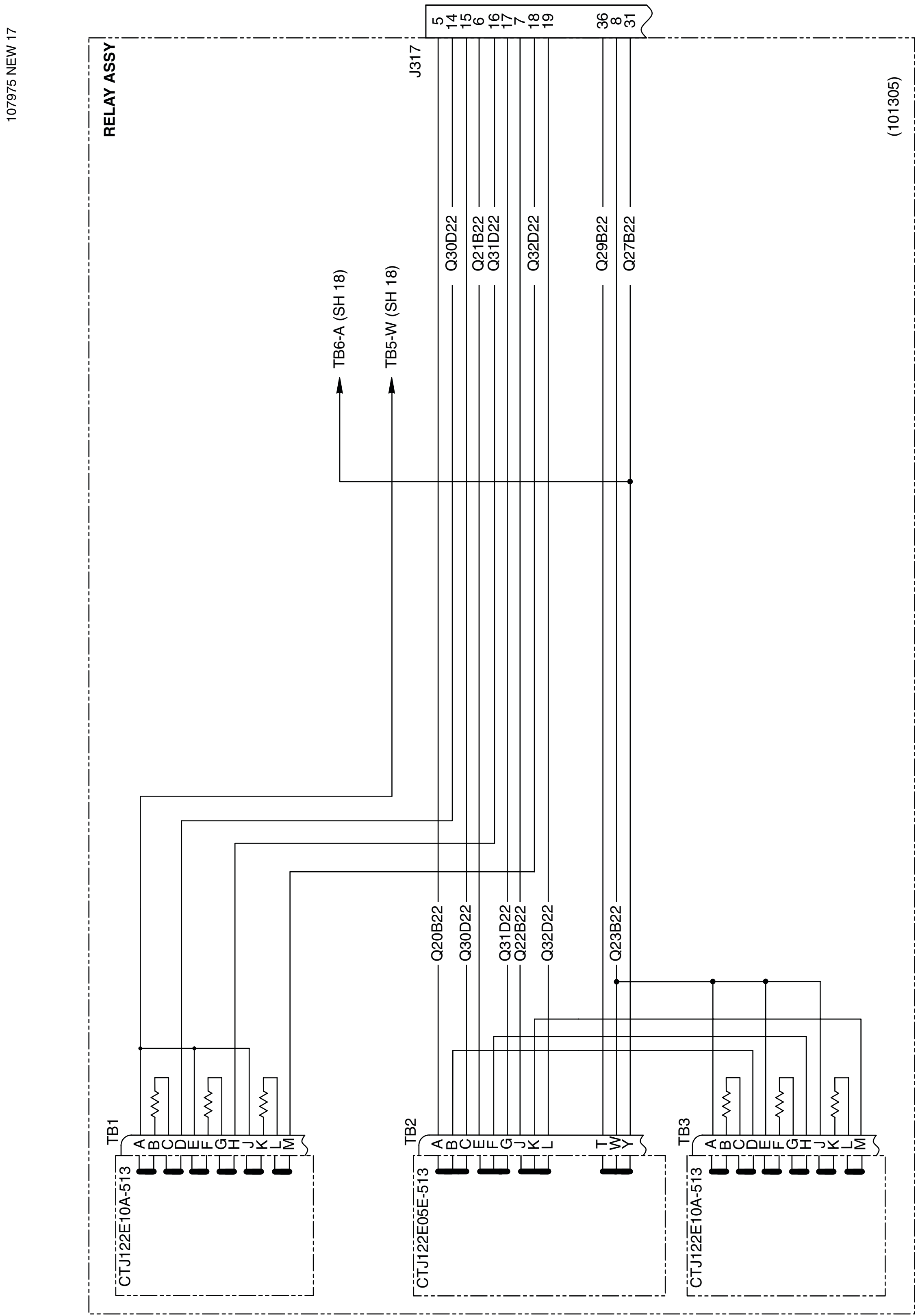
105552 NEW 39



Fuel Quantity Indicator
Figure 1 (Sheet 4 of 7)

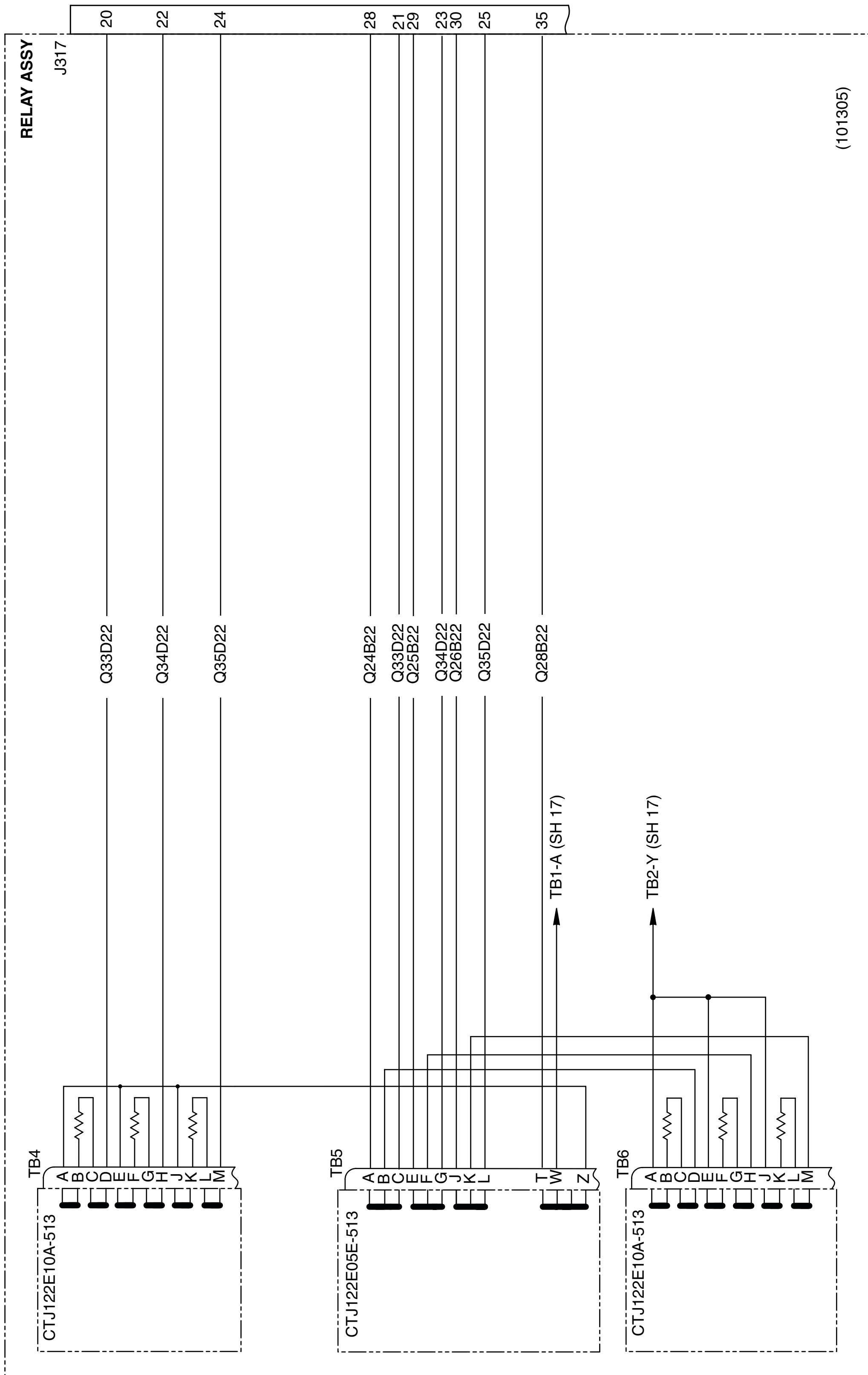


Fuel Quantity Indicator
Figure 1 (Sheet 5 of 7)



Fuel Quantity Indicator
Figure 1 (Sheet 6 of 7)

107975 NEW 18



Fuel Quantity Indicator
Figure 1 (Sheet 7 of 7)

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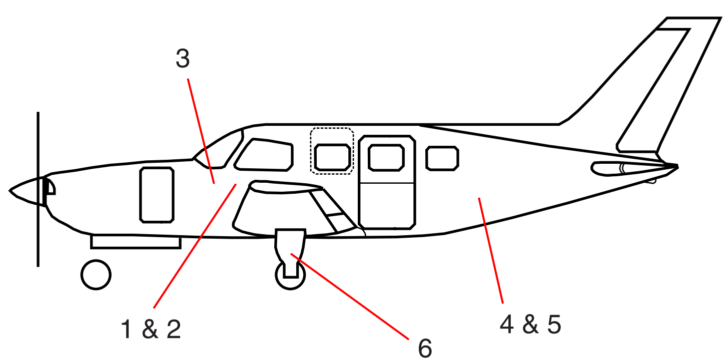
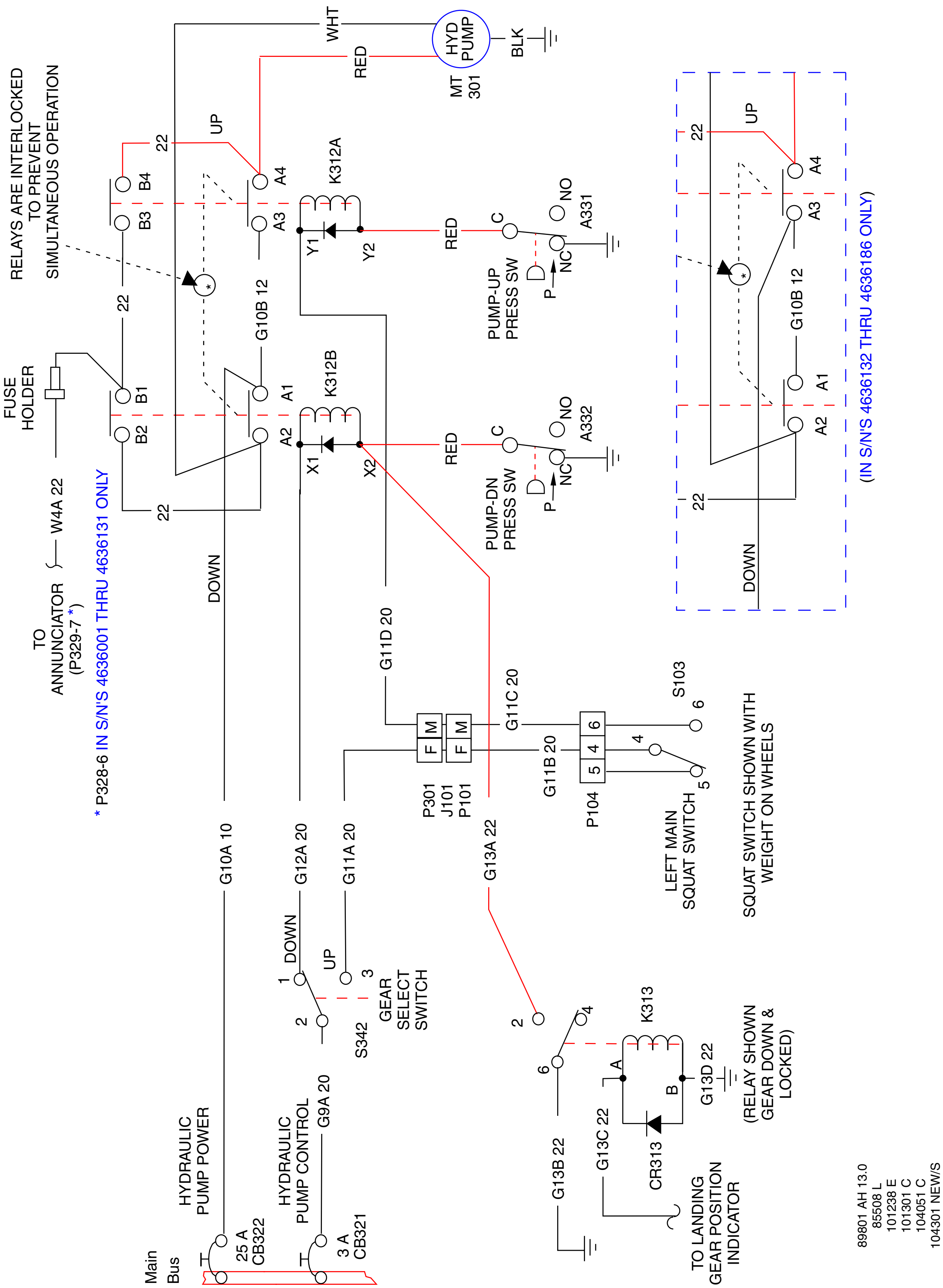
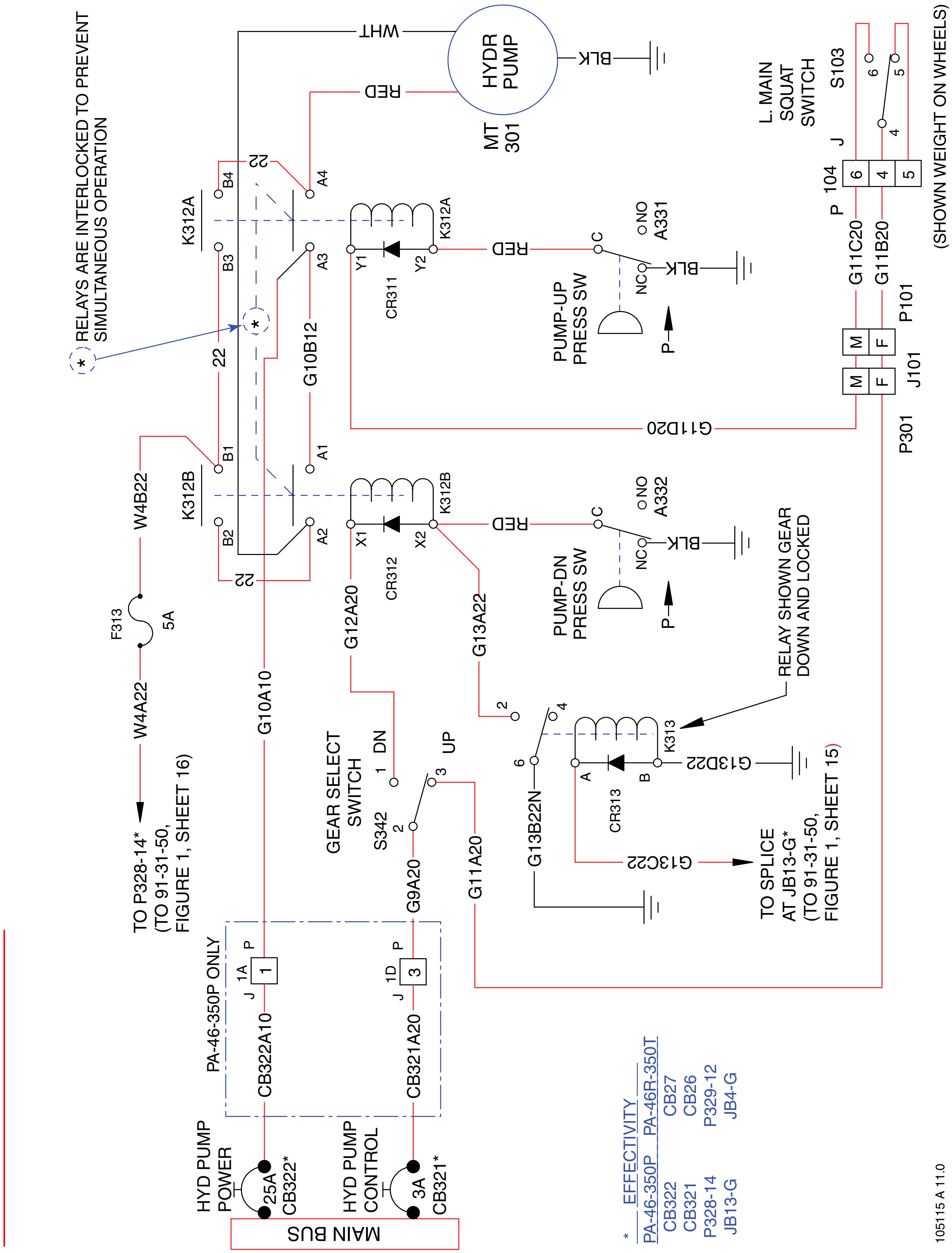
Item #	Designation	Description
1	CB322, CB27	Circuit Breaker - Hydraulic Pump Power (25 Amp)
2	CB321, CB26	Circuit Breaker - Hydraulic Pump Control (3 Amp)
3	S342	Gear Selector Switch
4	A332, A331	Pressure Switch Assembly
5	MT301	Hydraulic Pump
6	S103	Squat Switch Assembly



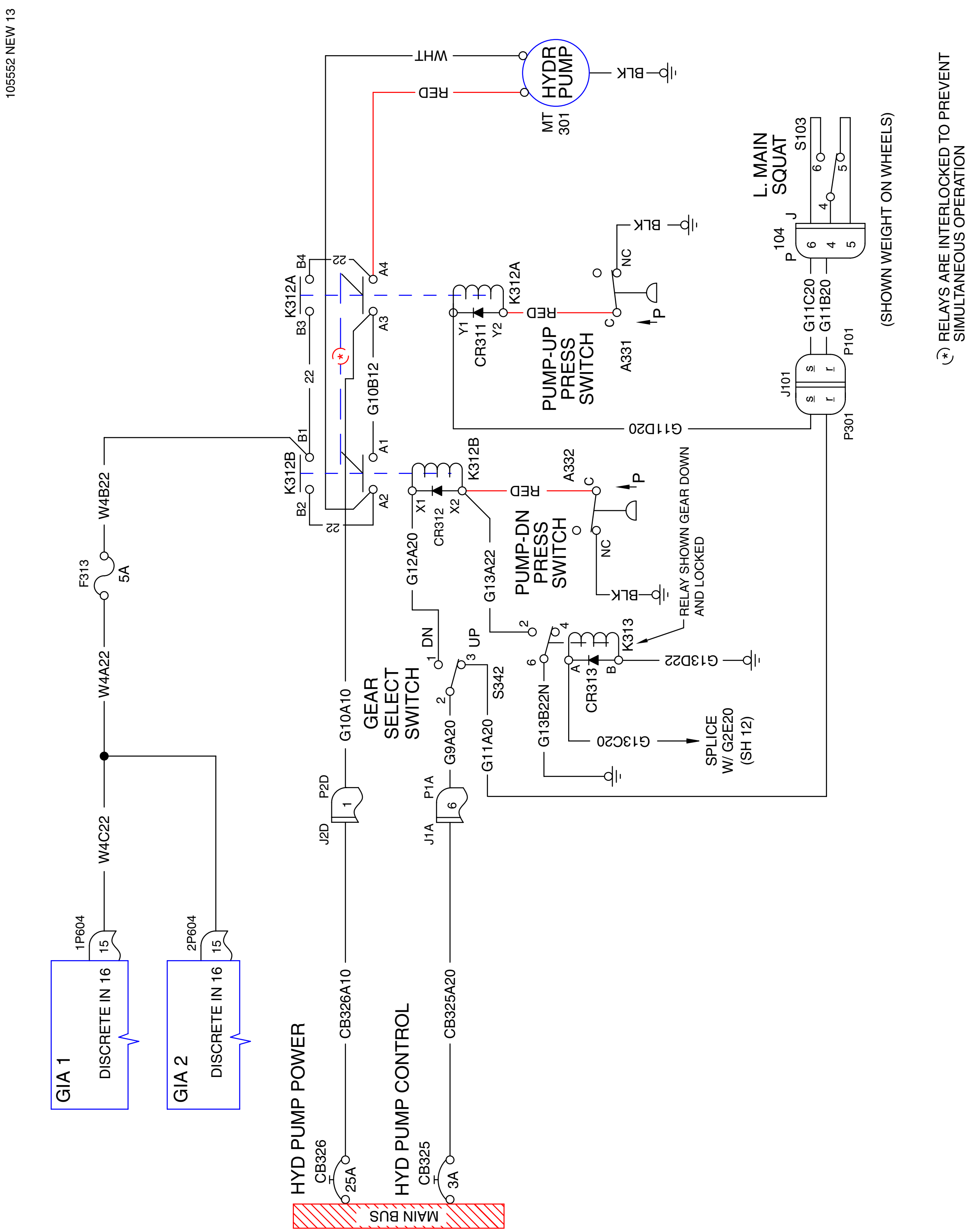
Hydraulic Pump
Figure 1 (Sheet 1 of 5)



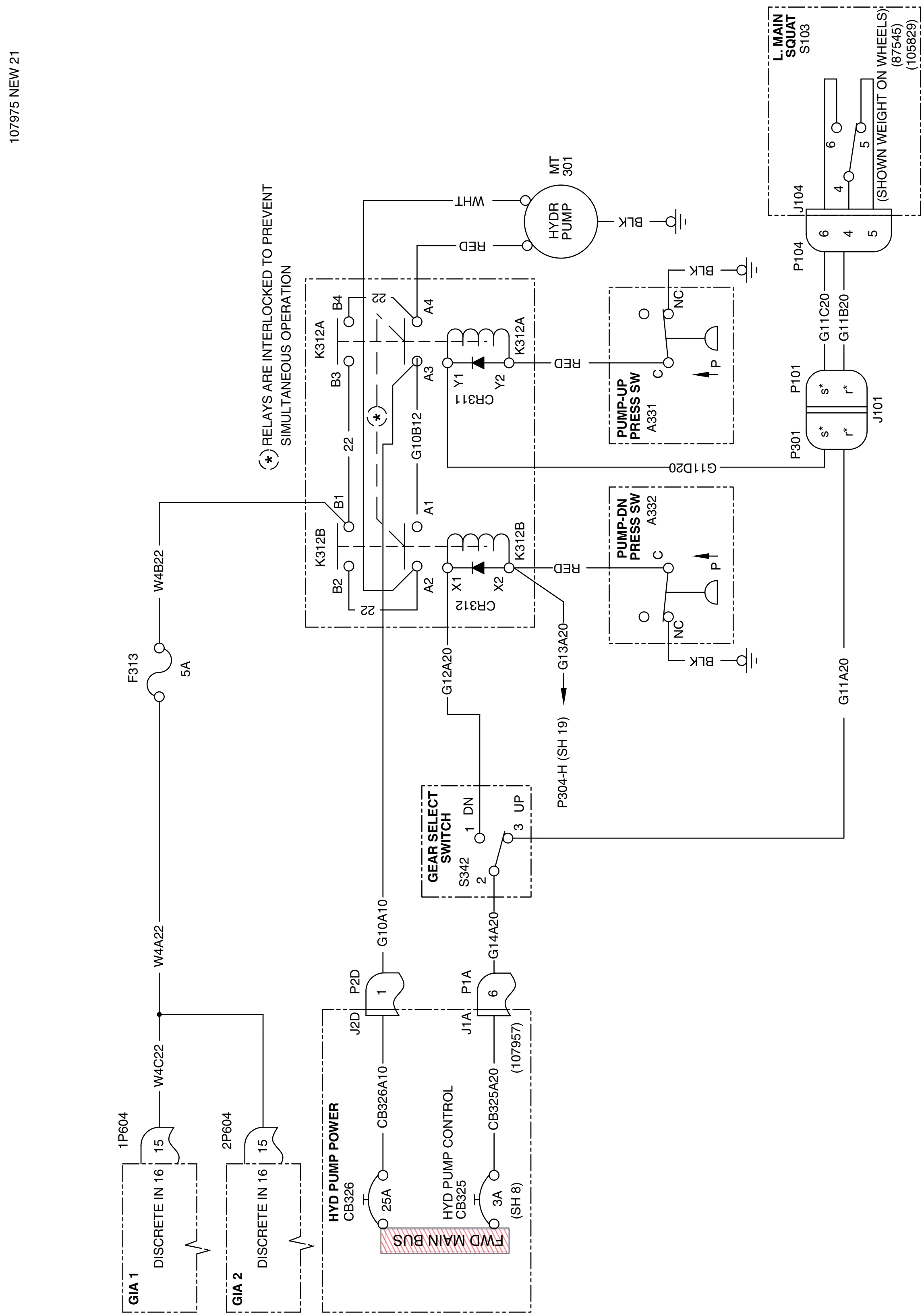
Hydraulic Pump
Figure 1 (Sheet 2 of 5)



Hydraulic Pump
Figure 1 (Sheet 3 of 5)



Hydraulic Pump
Figure 1 (Sheet 4 of 5)



Hydraulic Pump
Figure 1 (Sheet 5 of 5)

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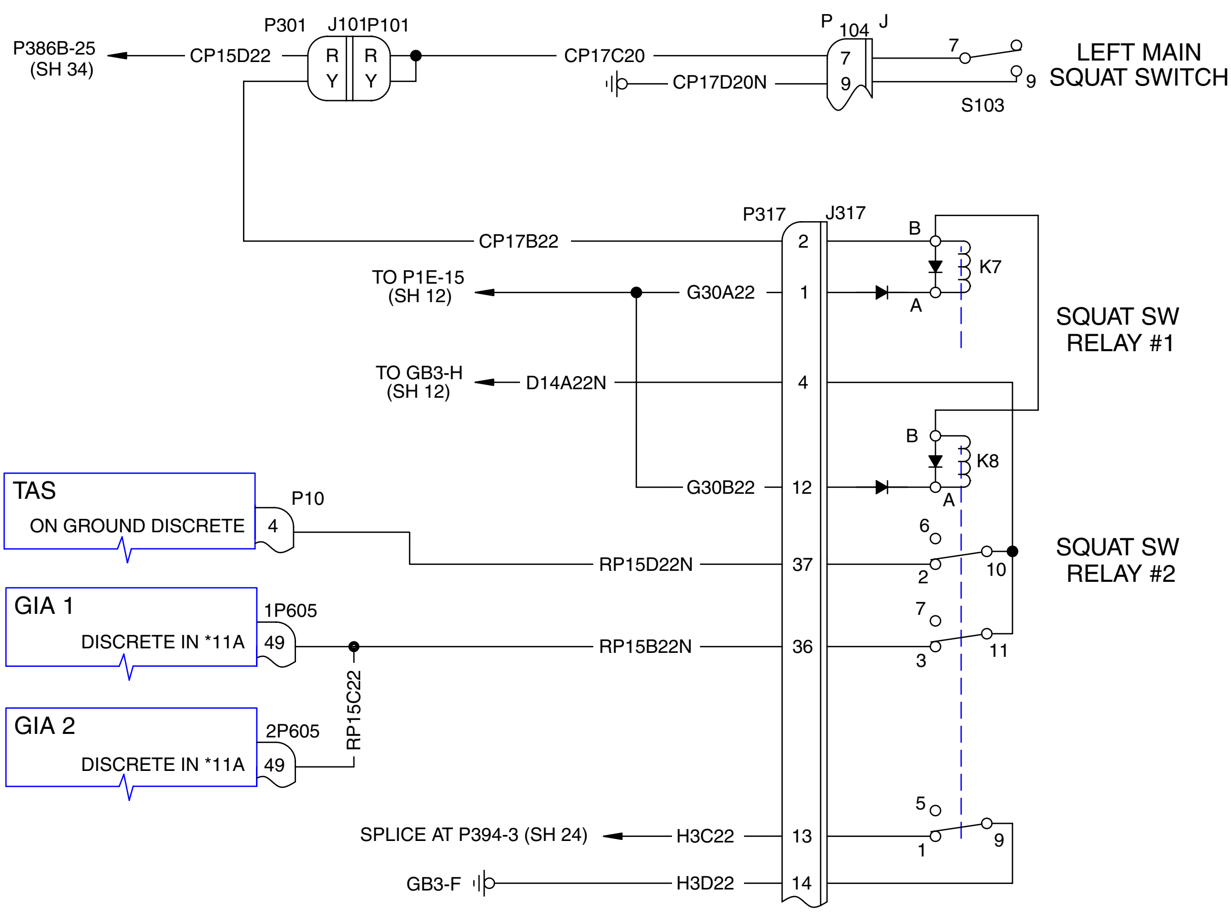
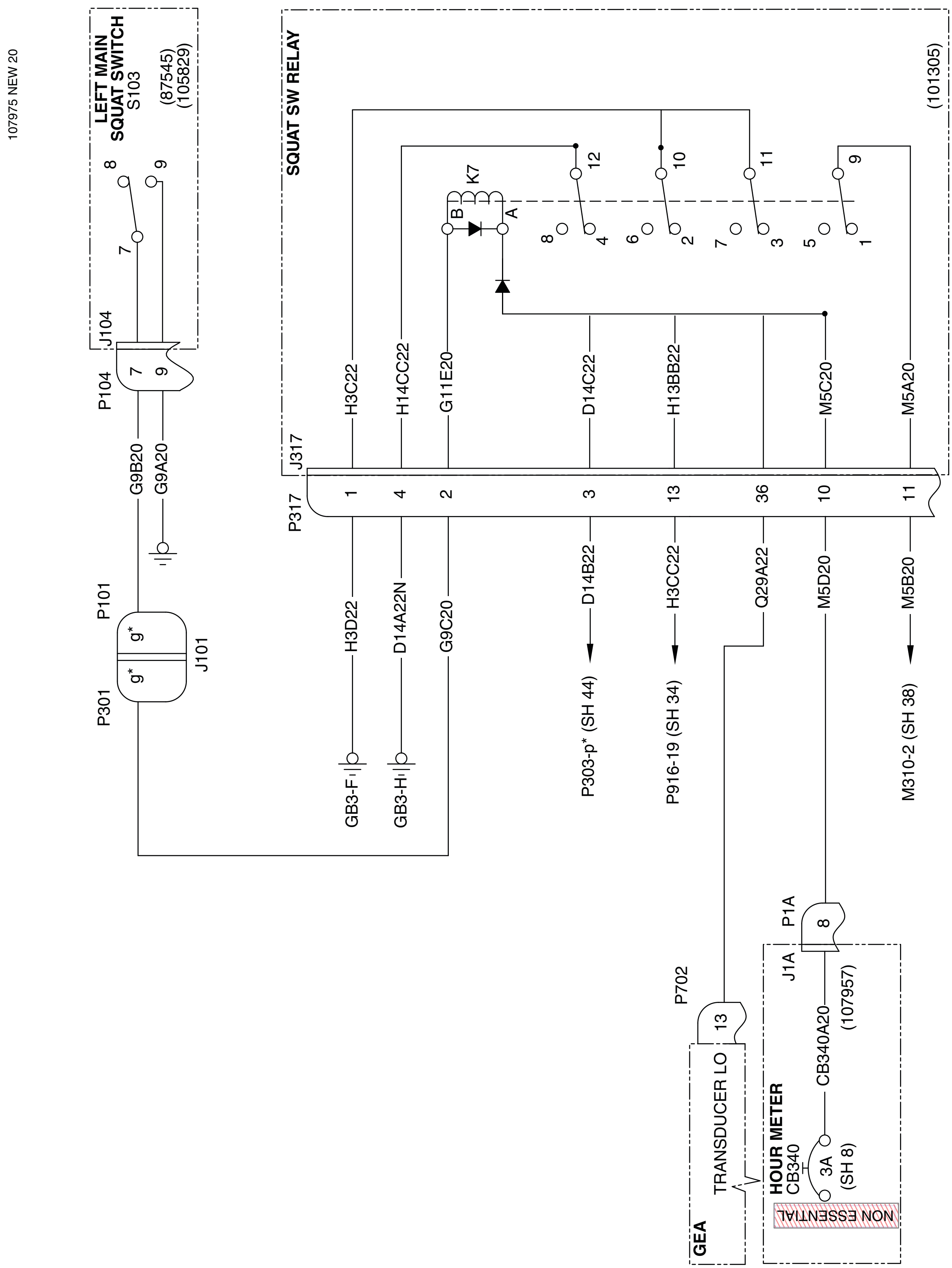


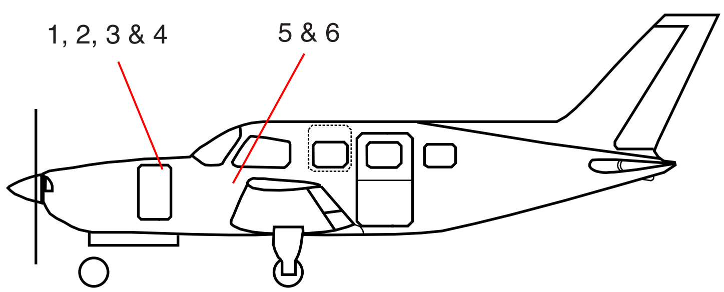
Squat Switch Relays
Figure 2 (Sheet 1 of 2)



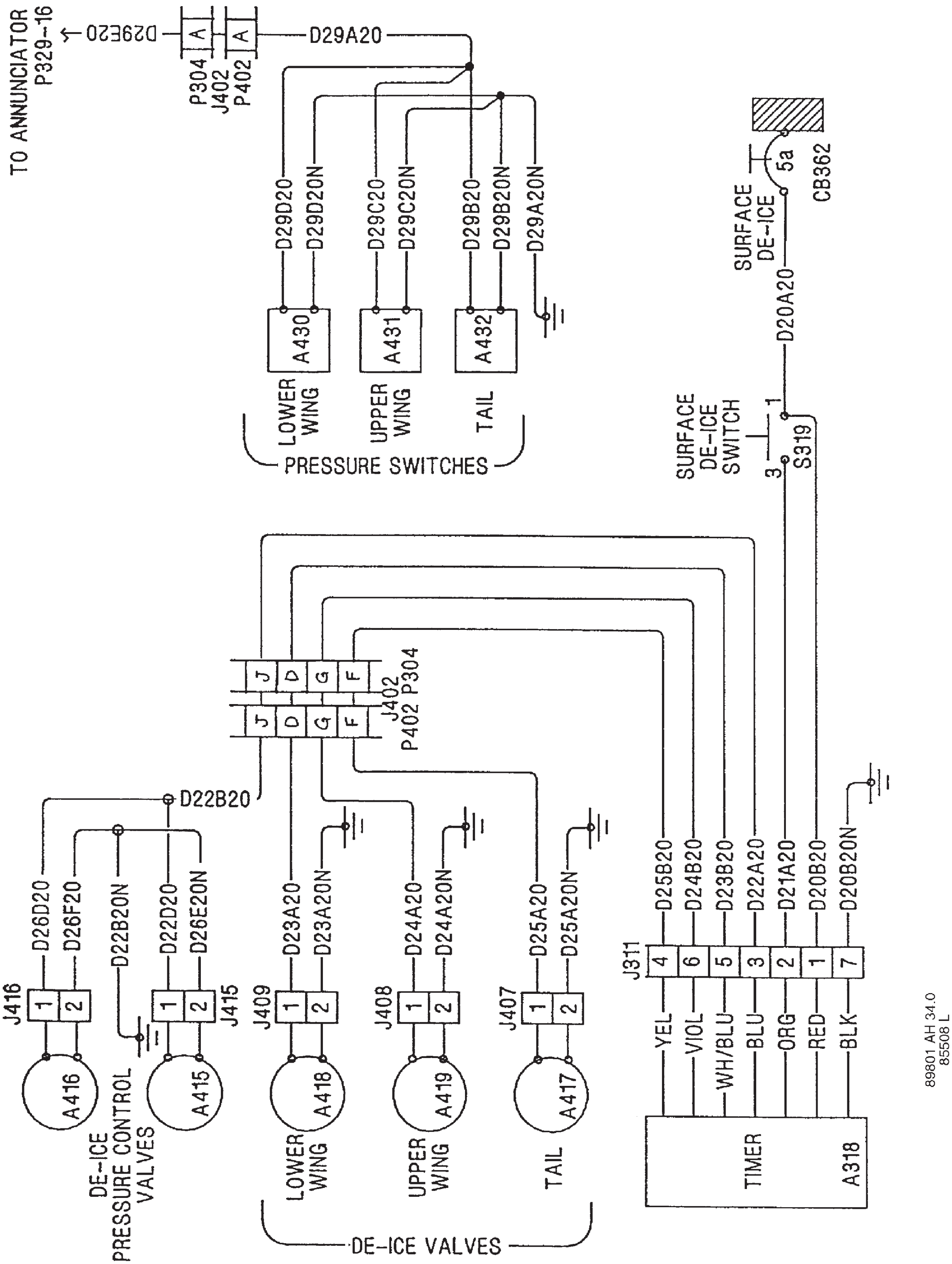
Squat Switch Relays
Figure 2 (Sheet 2 of 2)

9129108

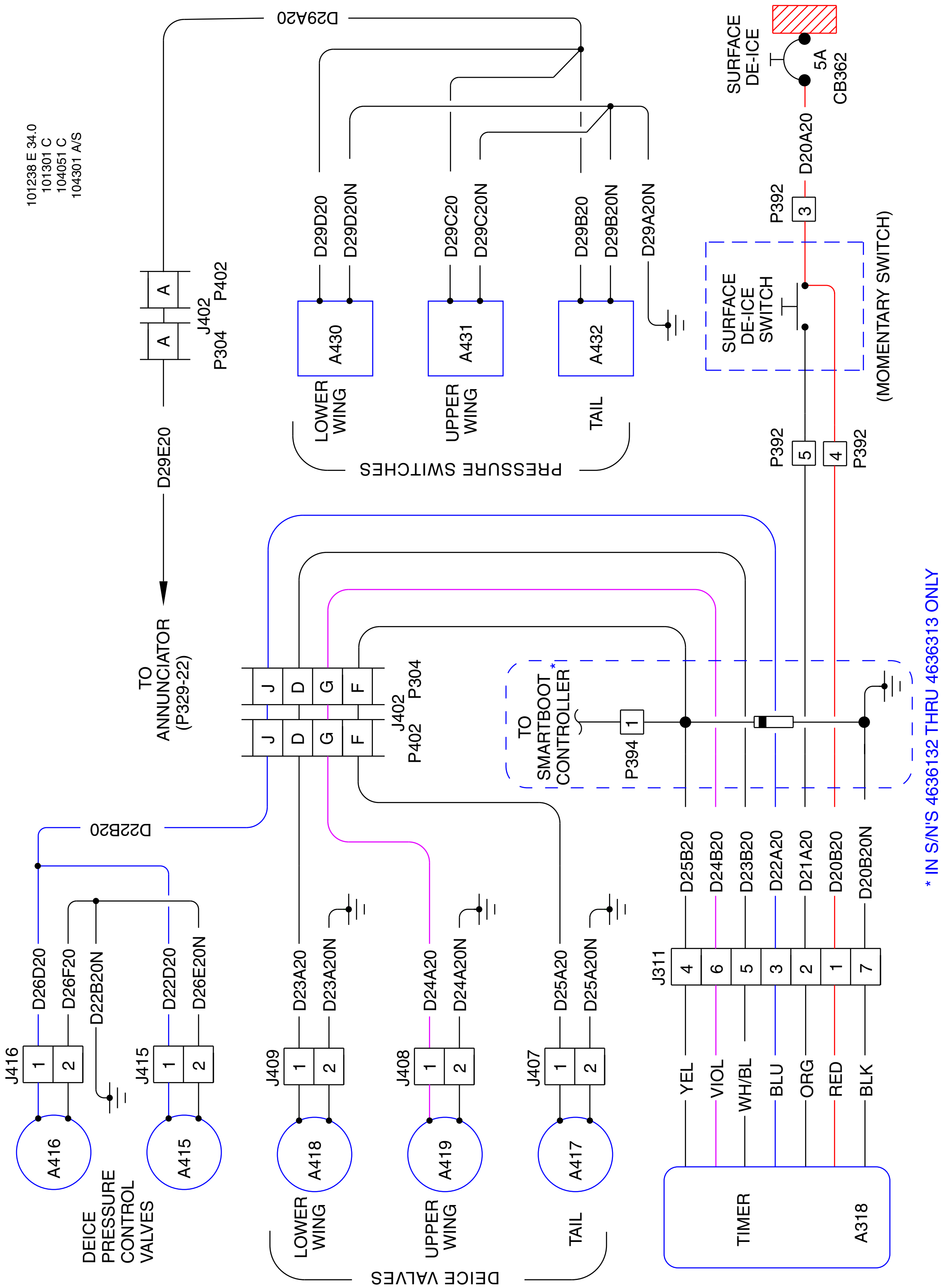
Item #	Designation	Description
1	A318	Timer Assembly
2	A415, A416	Pressure Control Valve
3	A417, A418, A419	De-Ice Valve
4	A430, A431, A432	Pressure Switch
5	CB362, CB45	Circuit Breaker - Surface De-Ice (5 Amp)
6	S319	Switch



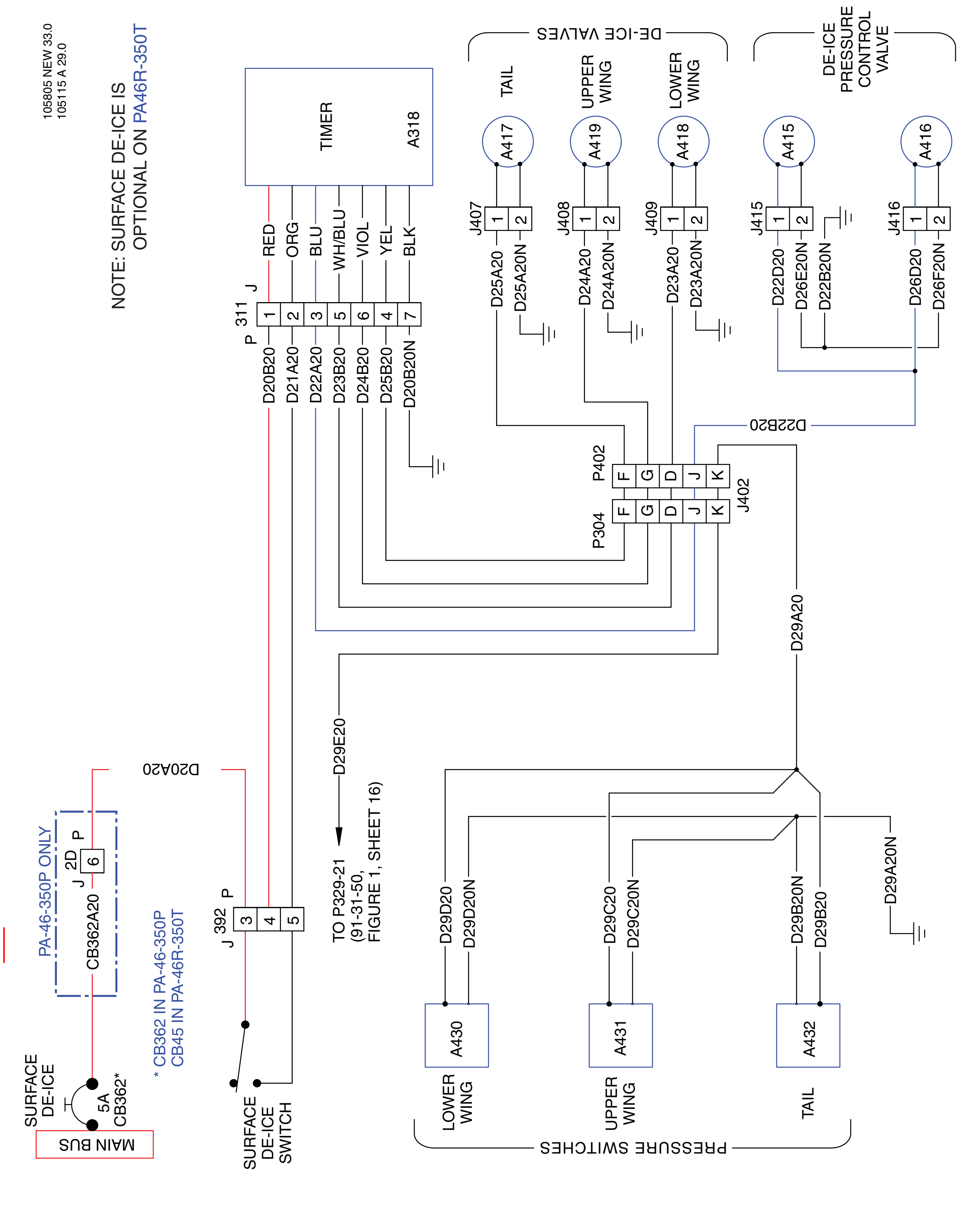
Surface Deice (Optional)
Figure 1 (Sheet 1 of 6)



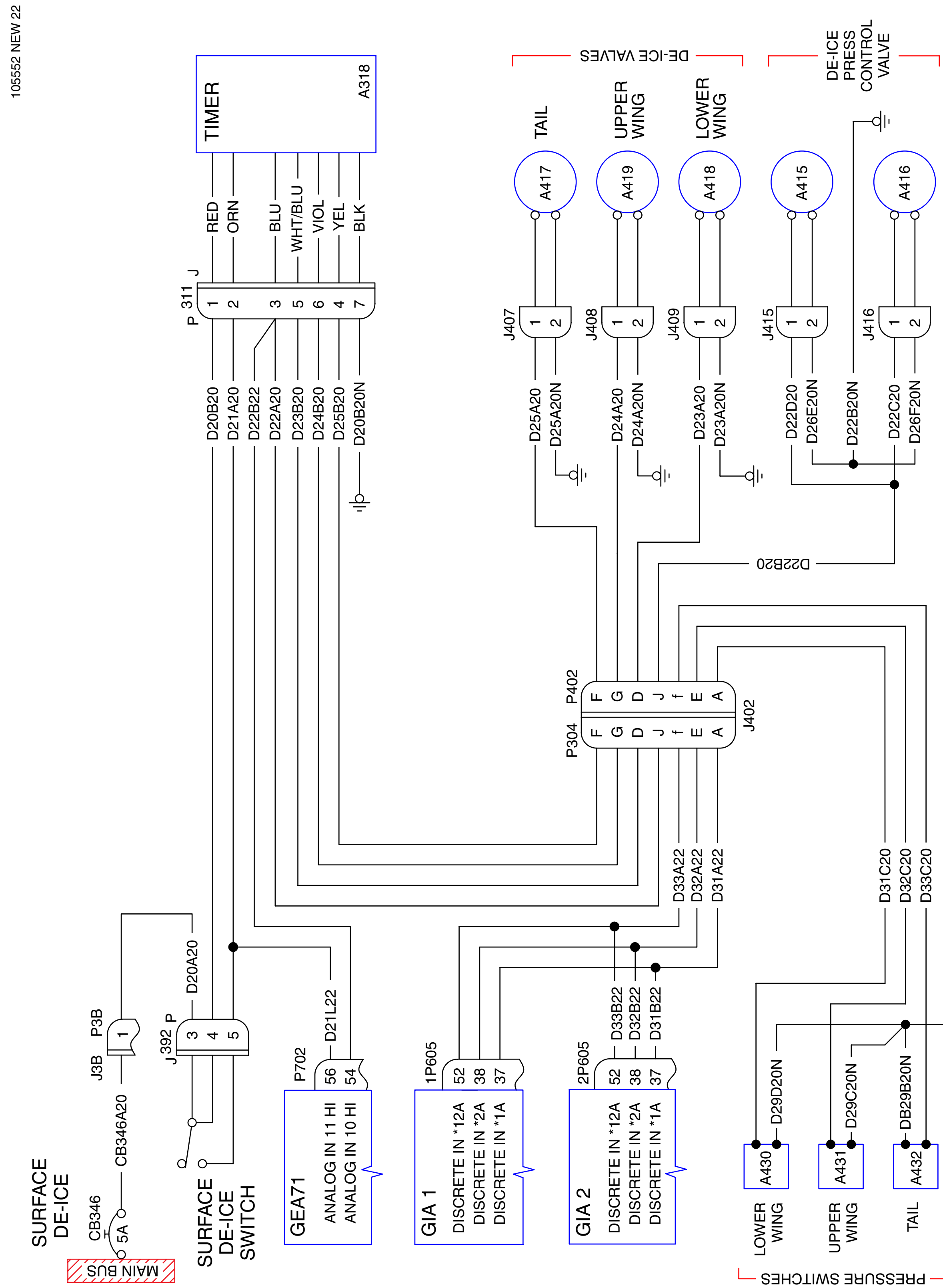
Surface Deice (Optional)
Figure 1 (Sheet 2 of 6)



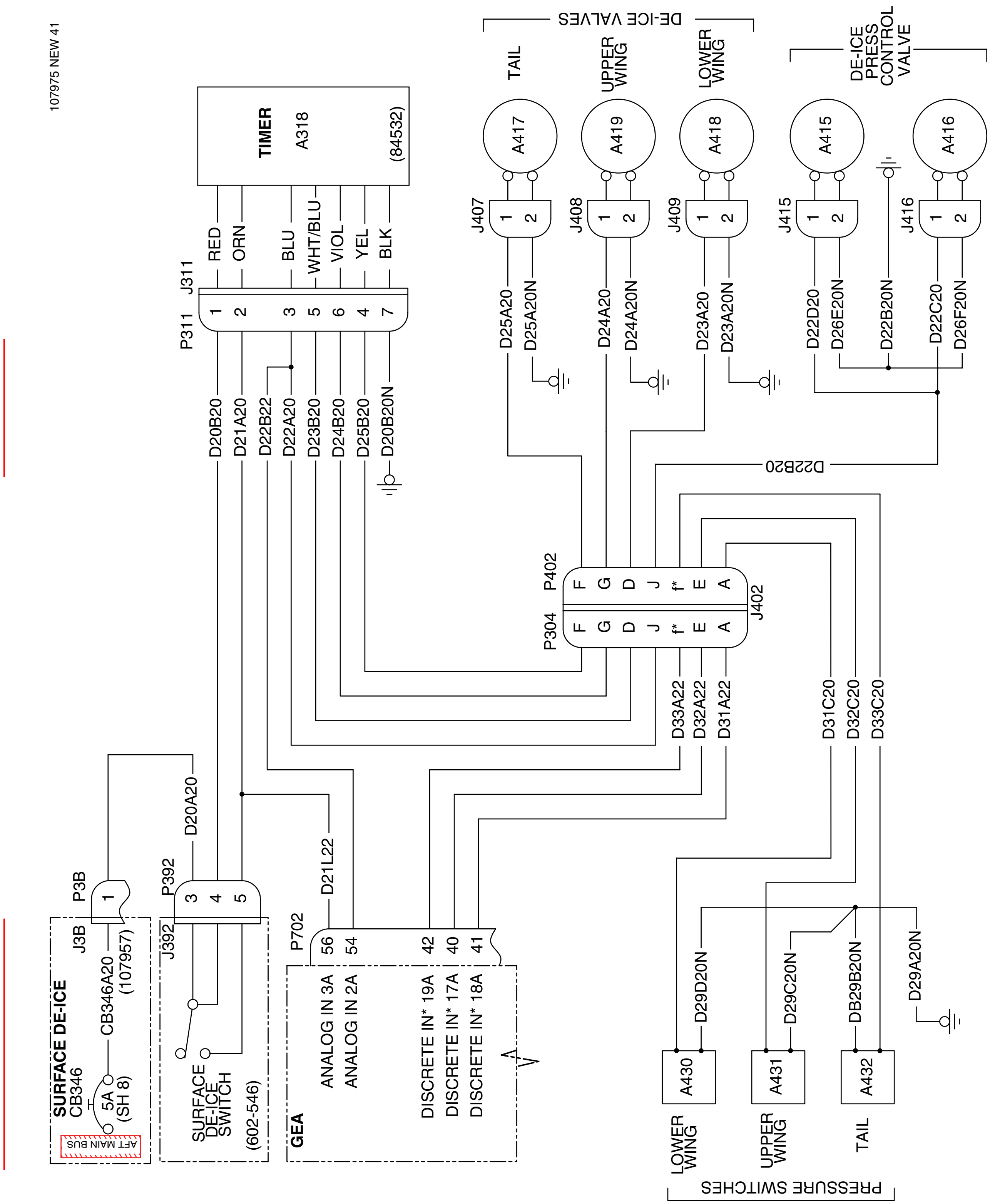
Surface Deice (Optional)
Figure 1 (Sheet 3 of 6)



Surface Deice (Optional)
Figure 1 (Sheet 4 of 6)

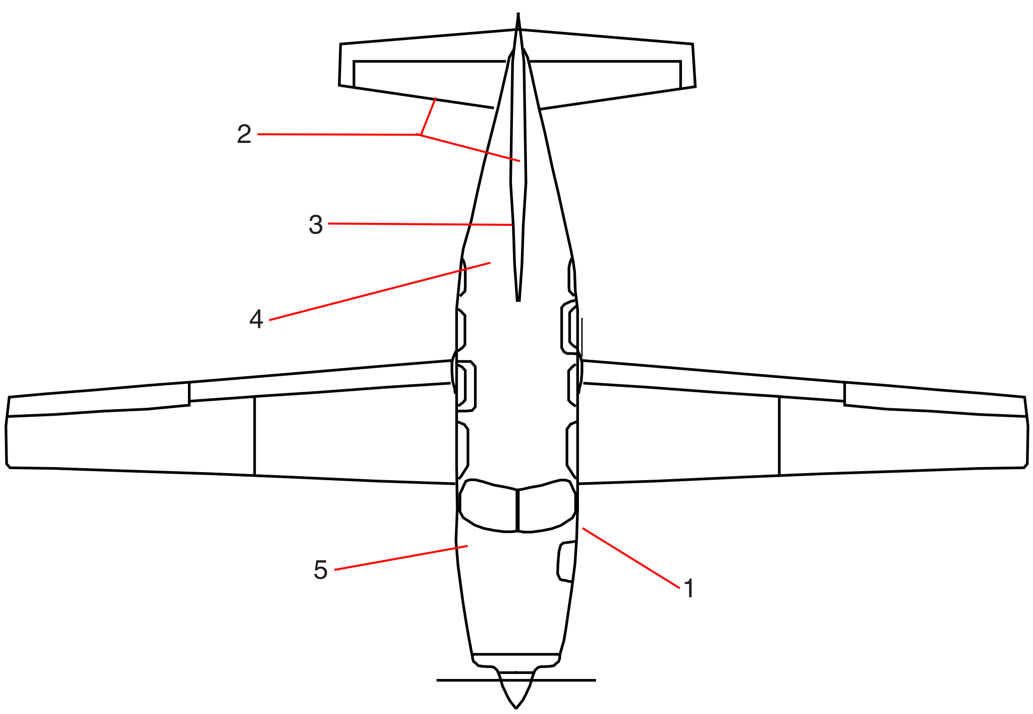


Surface Deice (Optional)
Figure 1 (Sheet 5 of 6)

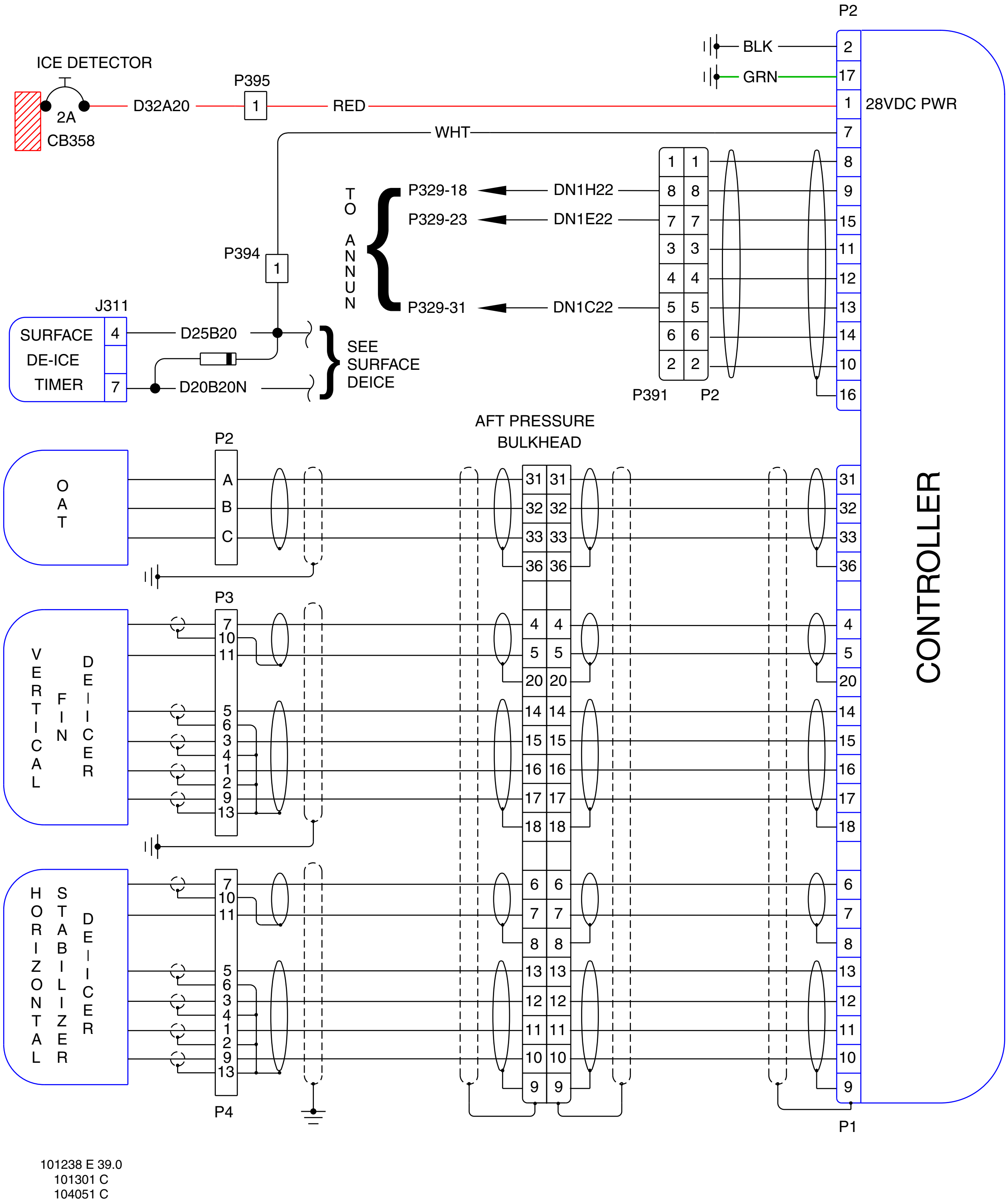


Surface Deice (Optional)
Figure 1 (Sheet 6 of 6)

Item #	Designation	Description
1	CB358	Circuit Breaker - Ice Detector (2 Amp)
2		De-icers with Integral Detectors
3		OAT Sensor
4		Controller
5	A318	Surface De-ice Timer



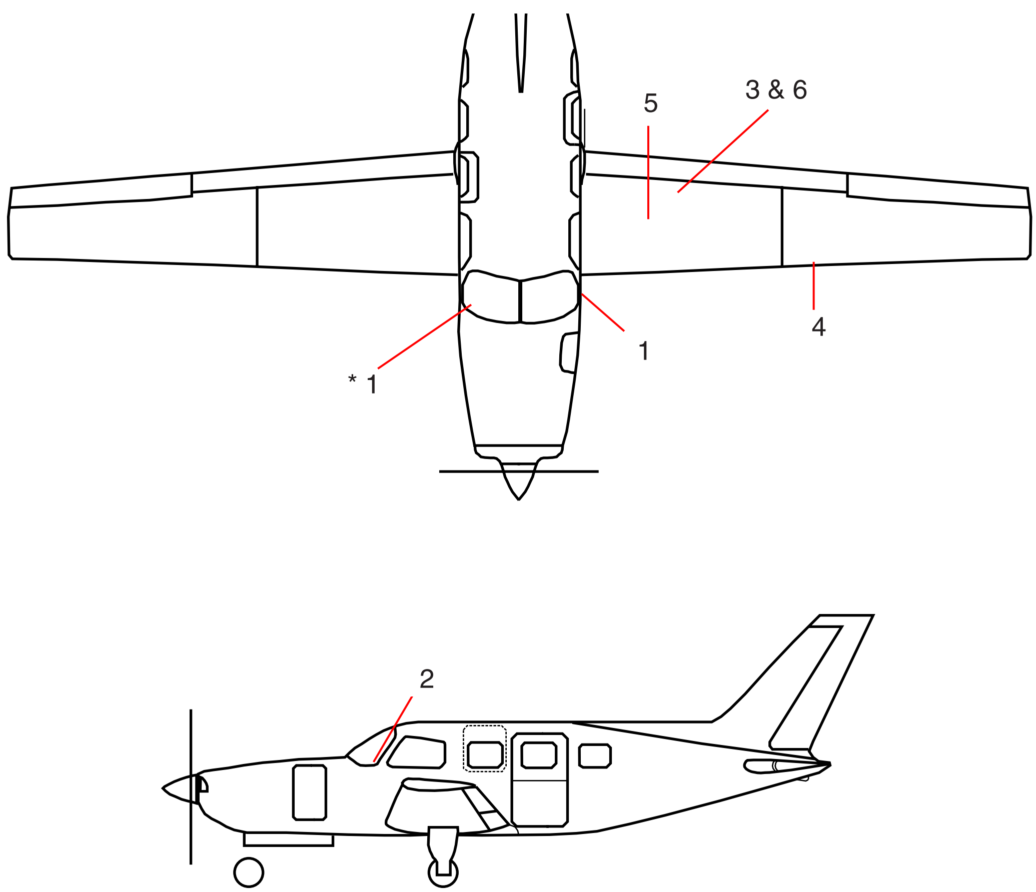
Ice Detection System (SMARTboots™) (Optional)
Figure 2 (Sheet 1 of 2)



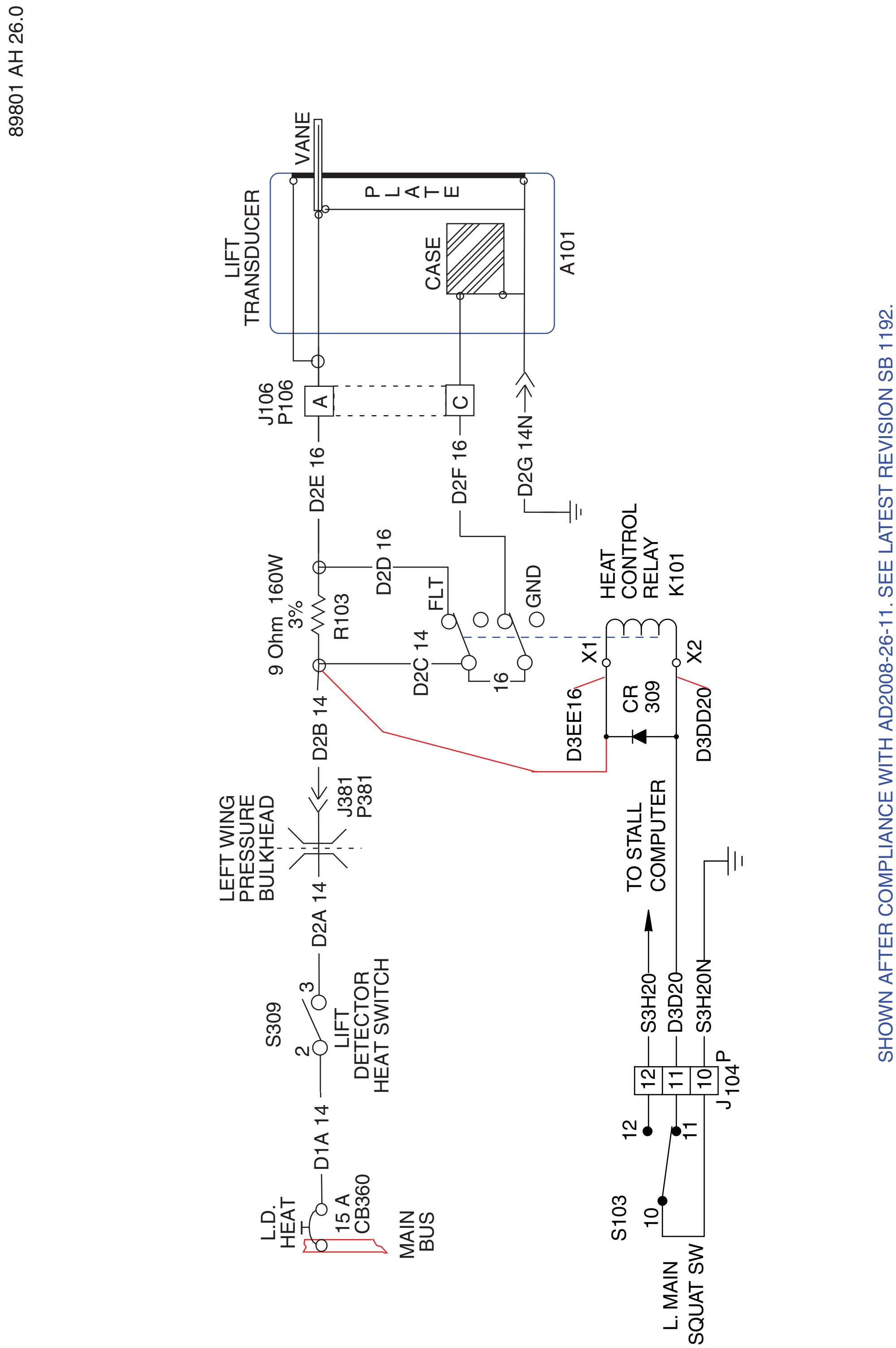
Ice Detection System (SMARTbootstm) (Optional)
Figure 2 (Sheet 2 of 2)

Item #	Designation	Description
1	CB360, CB42*	Circuit Breaker - Stall Heat - 15 Amp. (1))
	CB342	Circuit Breaker - Stall Heat - 25 Amp. (2))
2	S309	Stall Warn Heat Switch
3	R103	Resistor - Heated Stall Warn
4	A101	Lift Transducer
5	S101	Switch Assembly - Main Gear Uplock - Left
6	K101	Relay Assembly - Heated Stall Warning

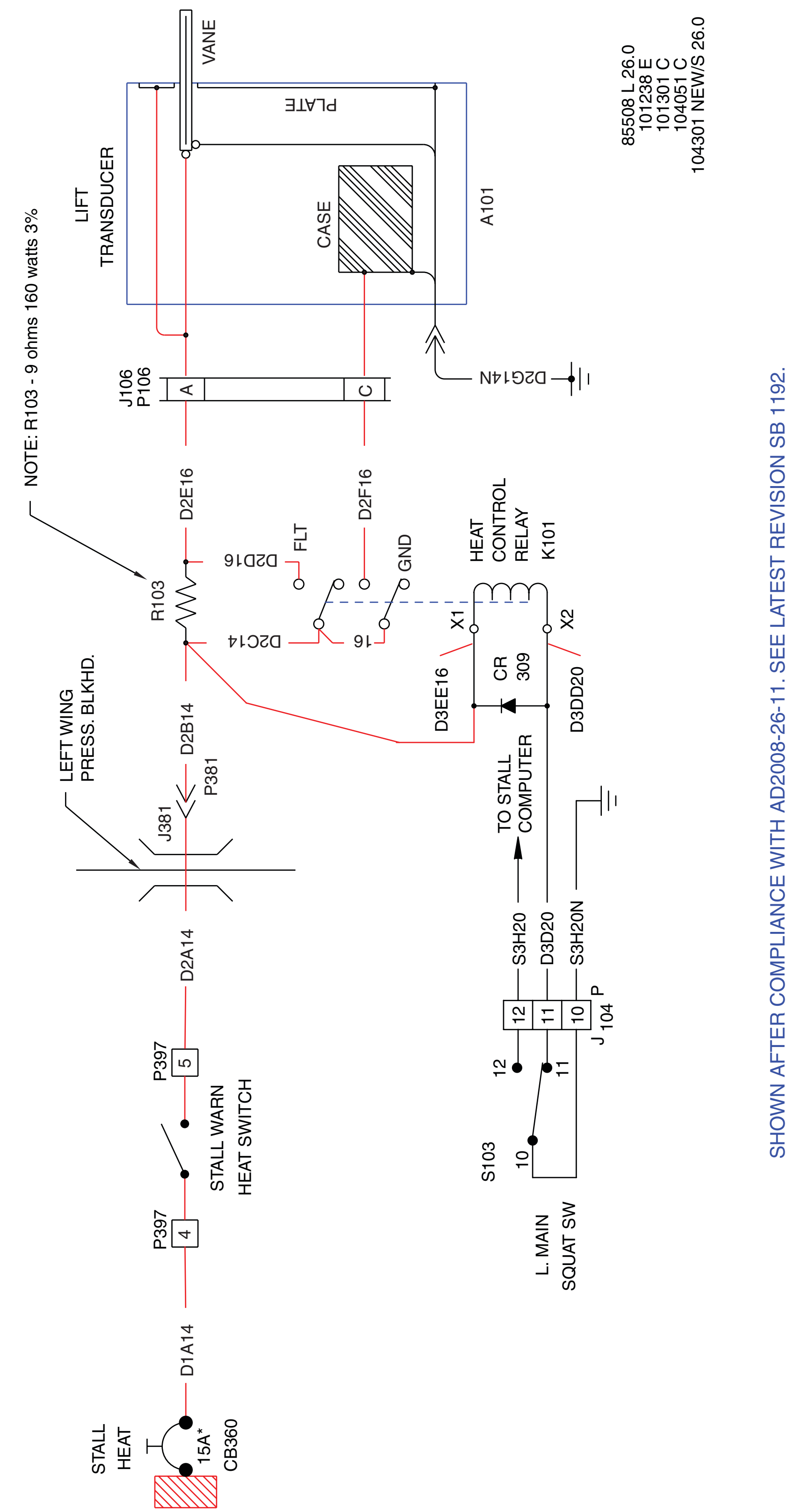
(1) S/N's 4636001–4636445 and 4692001–4692054 shown after compliance with AD2008-26-11. See latest revision SB 1192.
(2) With Garmin G1000 only.



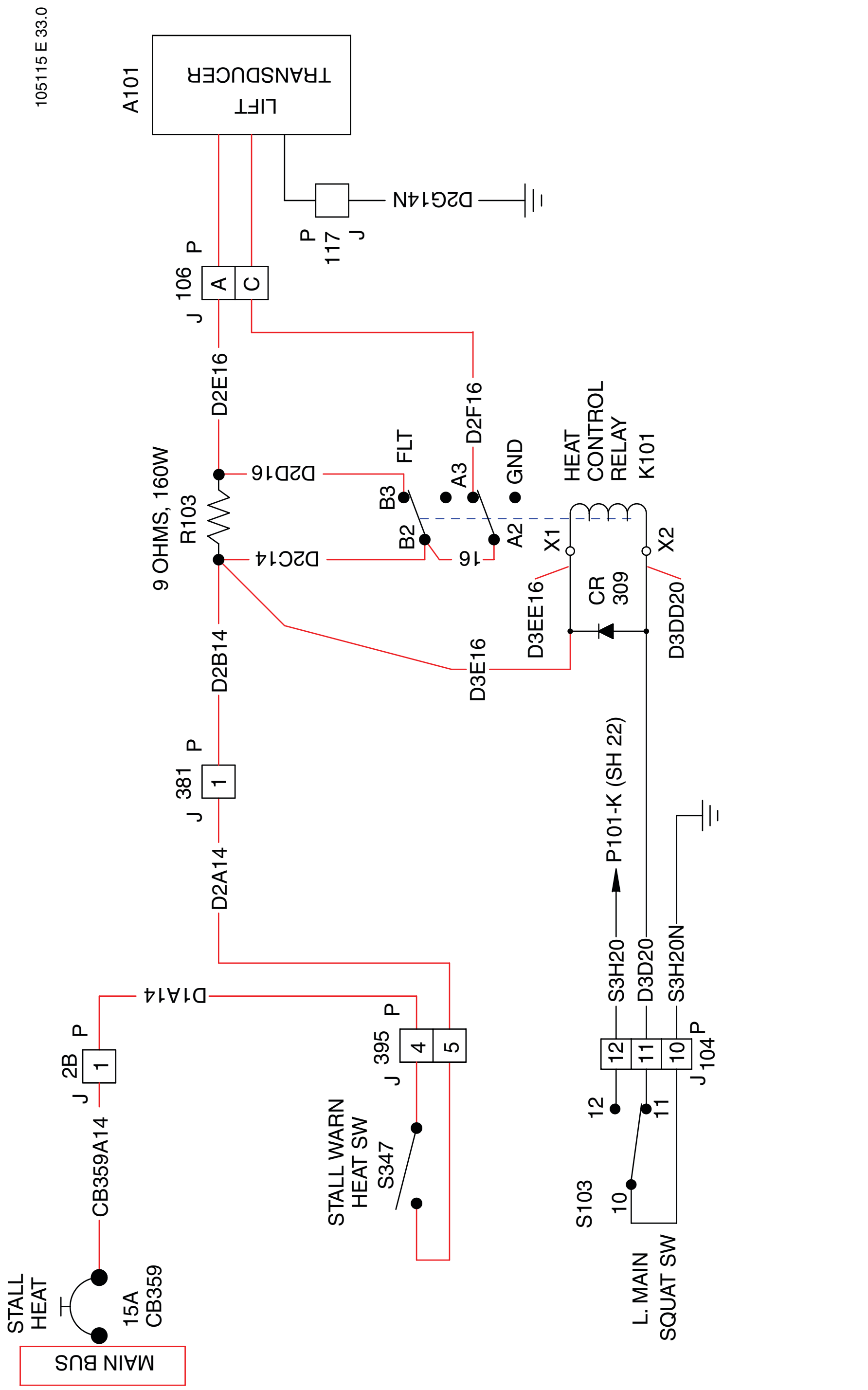
Lift Detector / Stall Warn Heat
Figure 1 (Sheet 1 of 9)



Lift Detector / Stall Warn Heat
Figure 1 (Sheet 2 of 9)



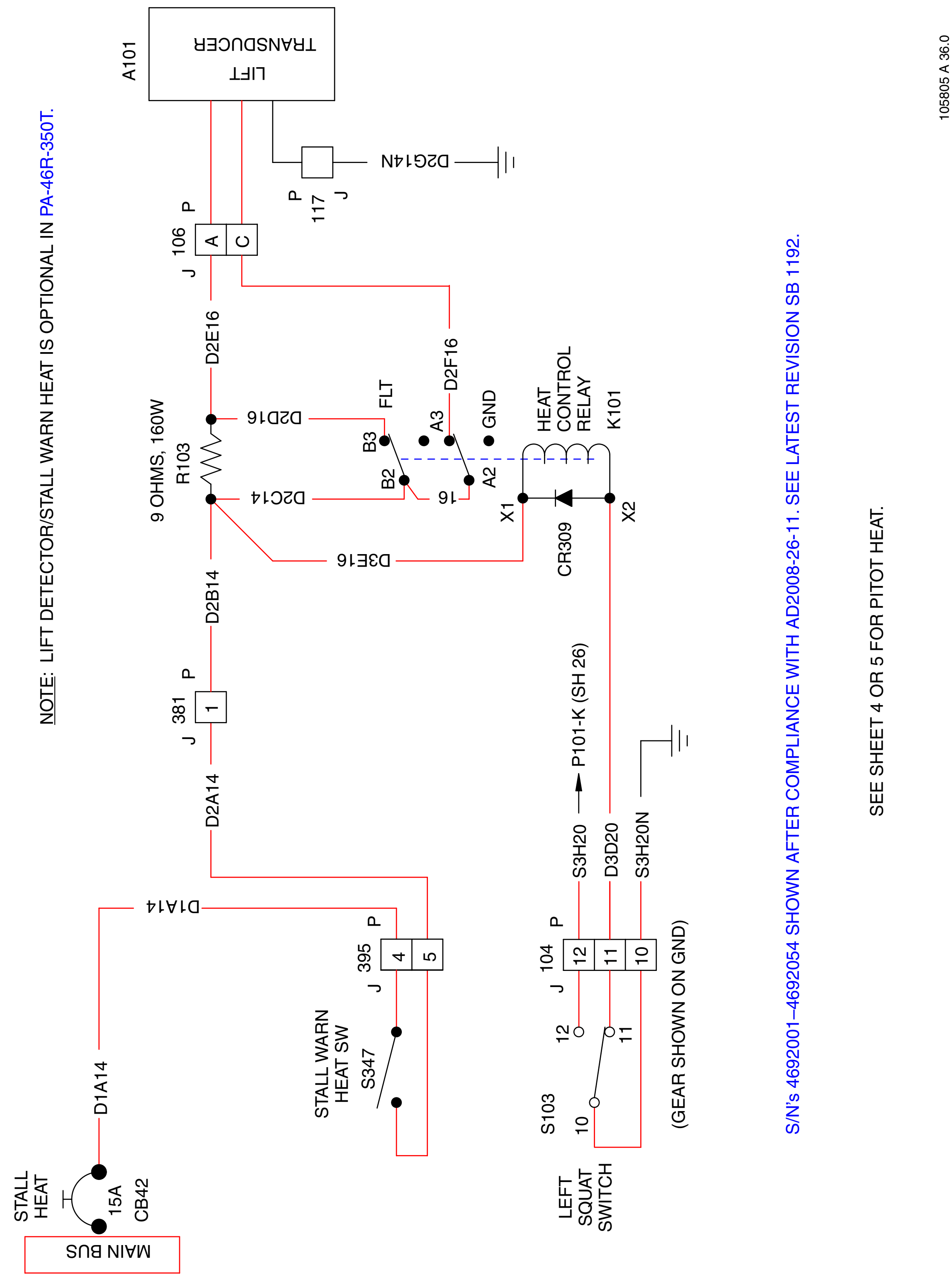
Lift Detector / Stall Warn Heat
Figure 1 (Sheet 3 of 9)



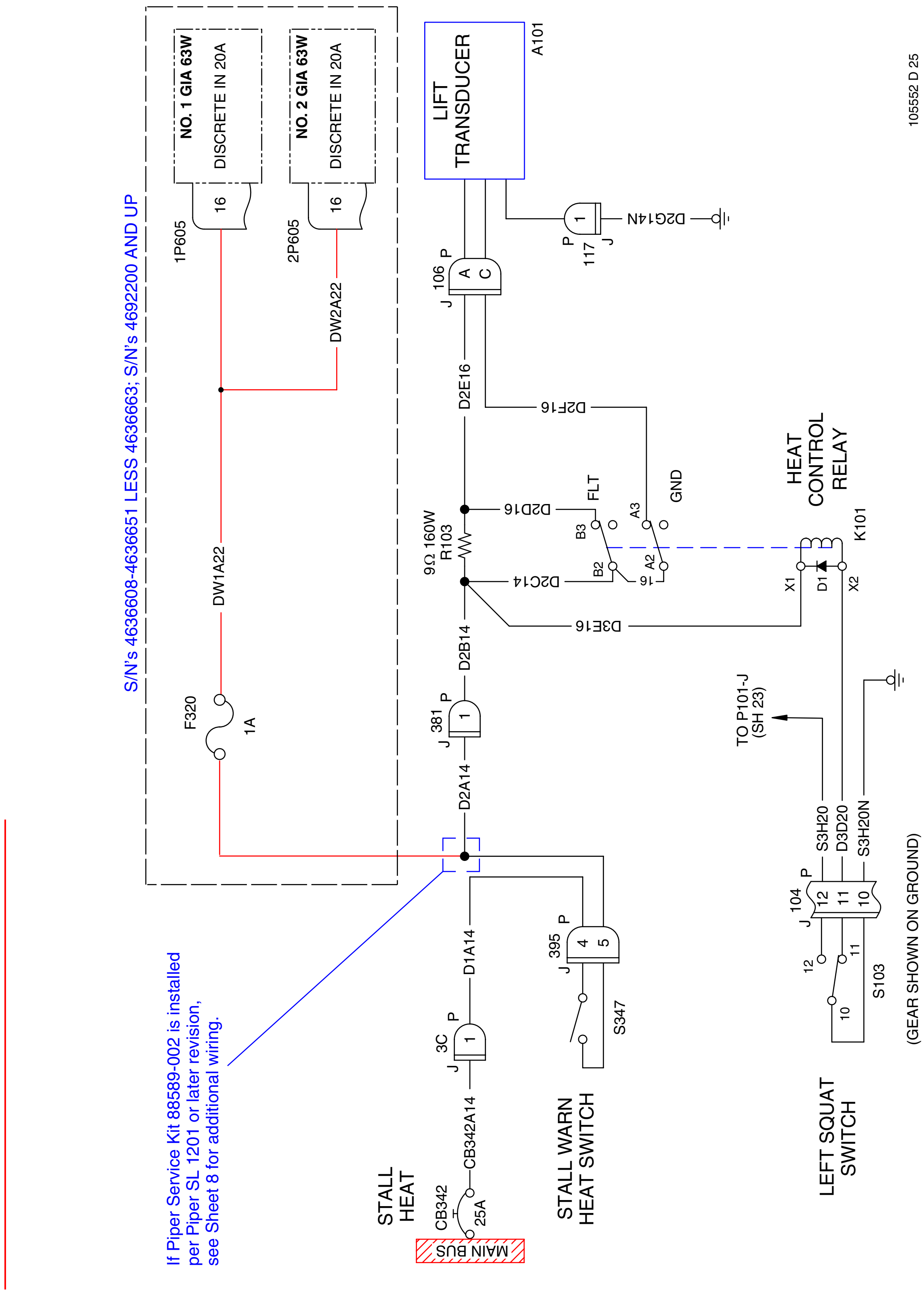
S/N's 4636001–4636445 SHOWN AFTER COMPLIANCE WITH AD2008-26-11. SEE LATEST REVISION SB 1192.

SEE SHEET 4 OR 5 FOR PITOT HEAT.

Lift Detector / Stall Warn Heat
Figure 1 (Sheet 4 of 9)

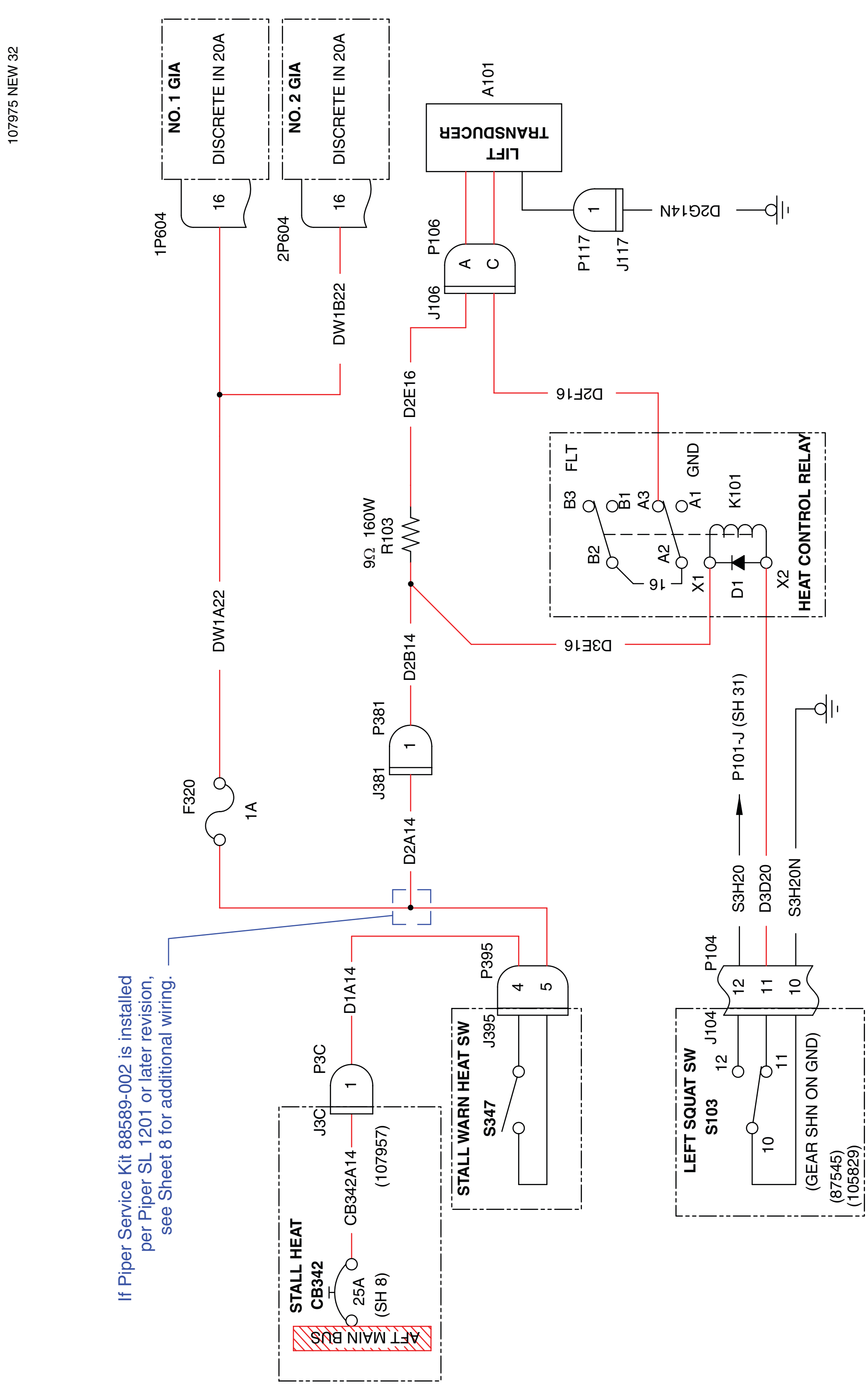


Lift Detector / Stall Warn Heat
Figure 1 (Sheet 5 of 9)

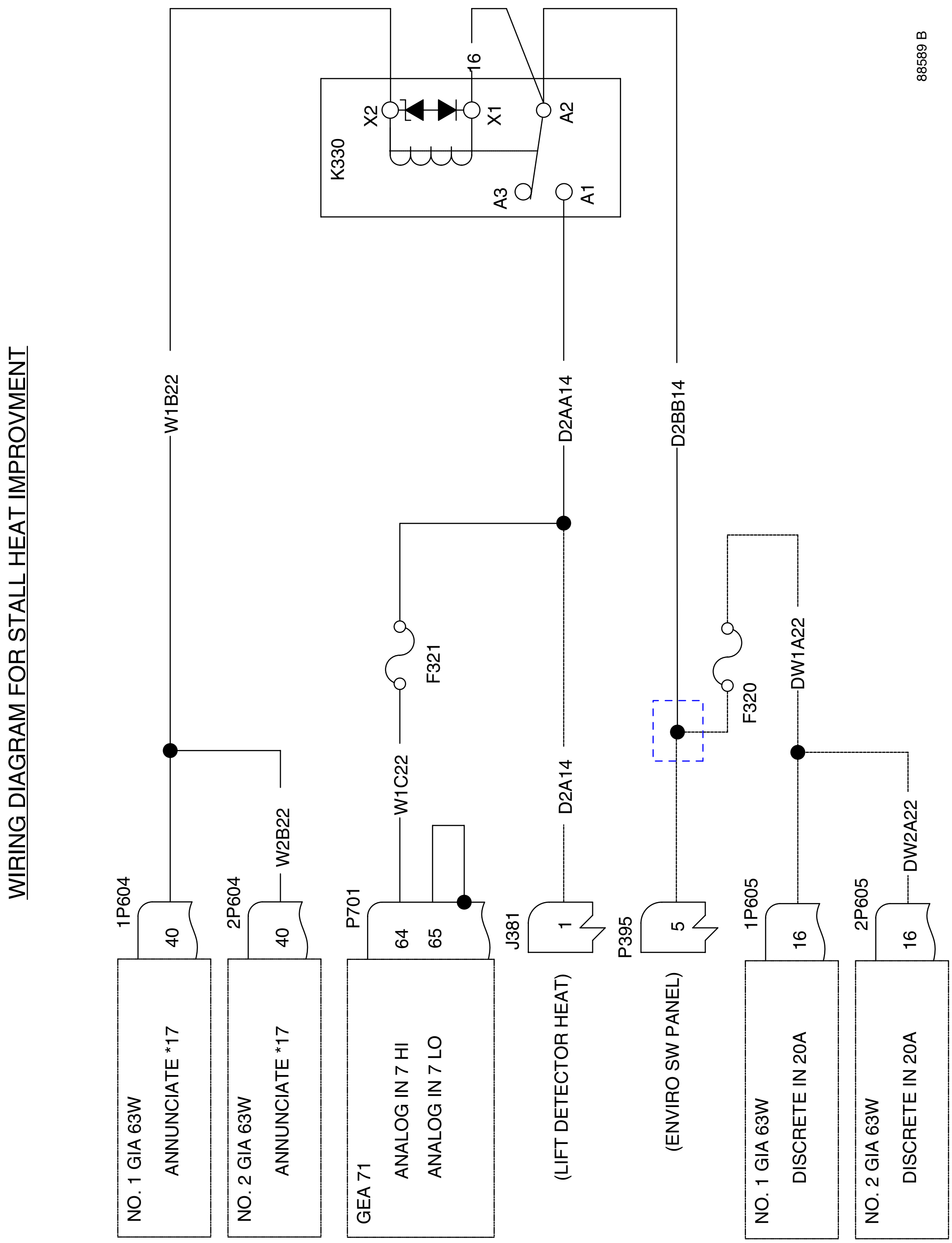


Lift Detector / Stall Warn Heat
Figure 1 (Sheet 6 of 9)

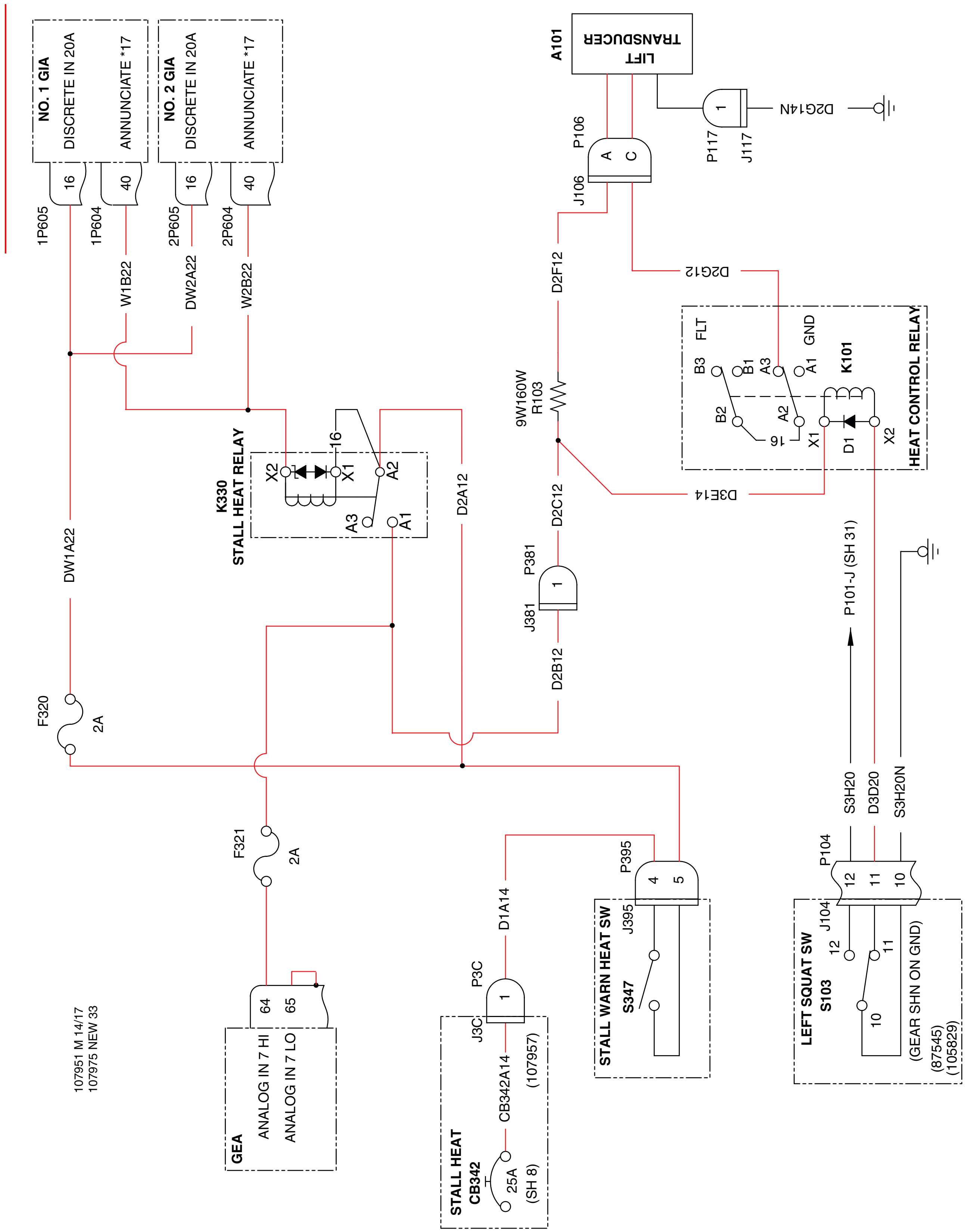
Effectivity
4636460, 4636463–4636651, less 4636481 and 4636633
4692134 and up, less 4692141, 4692149, 4692153



Lift Detector / Stall Warn Heat
Figure 1 (Sheet 7 of 9)



Lift Detector / Stall Warn Heat
Figure 1 (Sheet 8 of 9)



Lift Detector / Stall Warn Heat
Figure 1 (Sheet 9 of 9)

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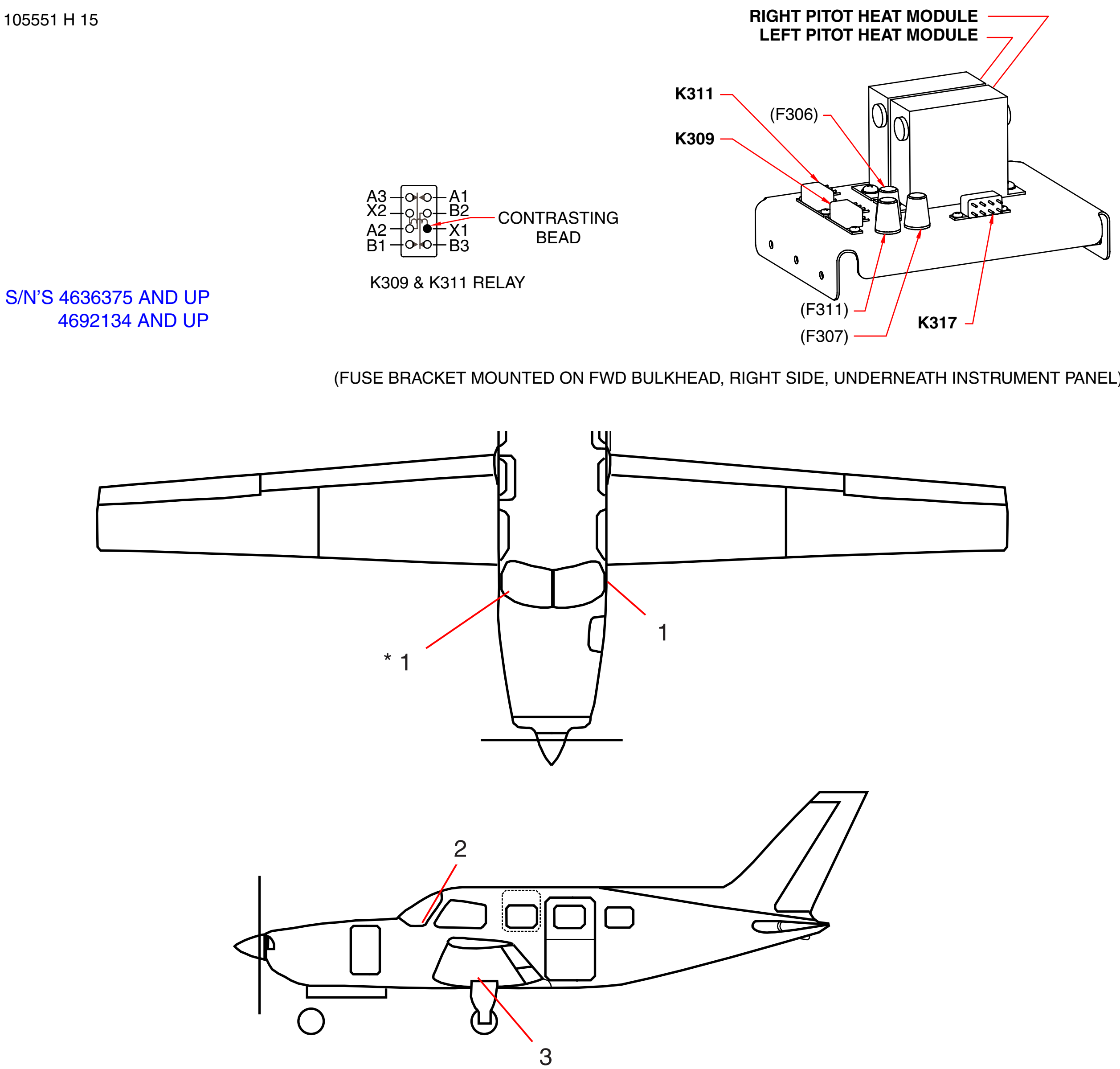
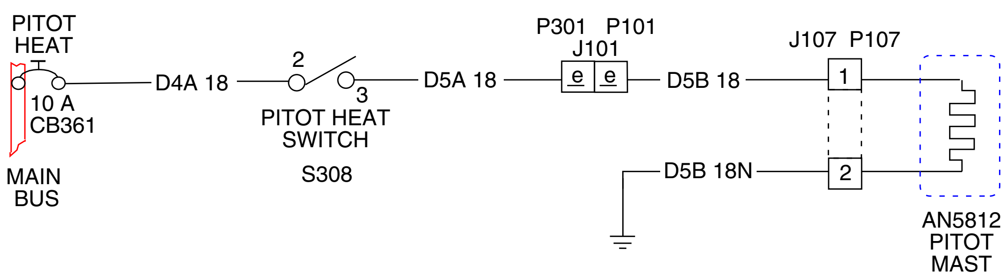
Item #	Designation	Description
1	CB361, CB43*	Circuit Breaker - Pitot Heat (10 Amp)
2	S308	Pitot Heat Switch
3	AN5812	Pitot Head Assembly - Heated (Left and Right Wings with Dual PFD Avidyne Entegra)

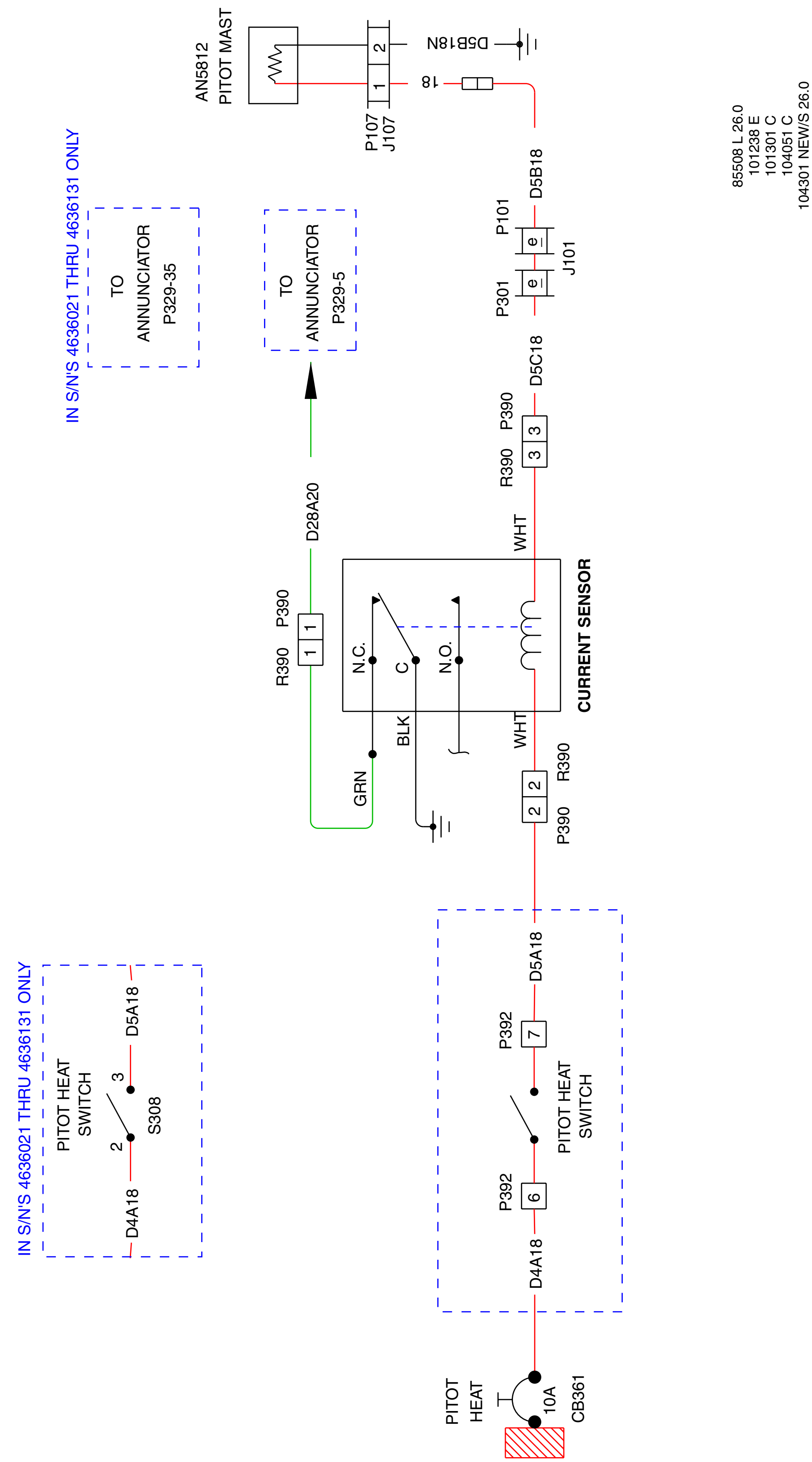


Pitot Heat
Figure 2 (Sheet 1 of 10)

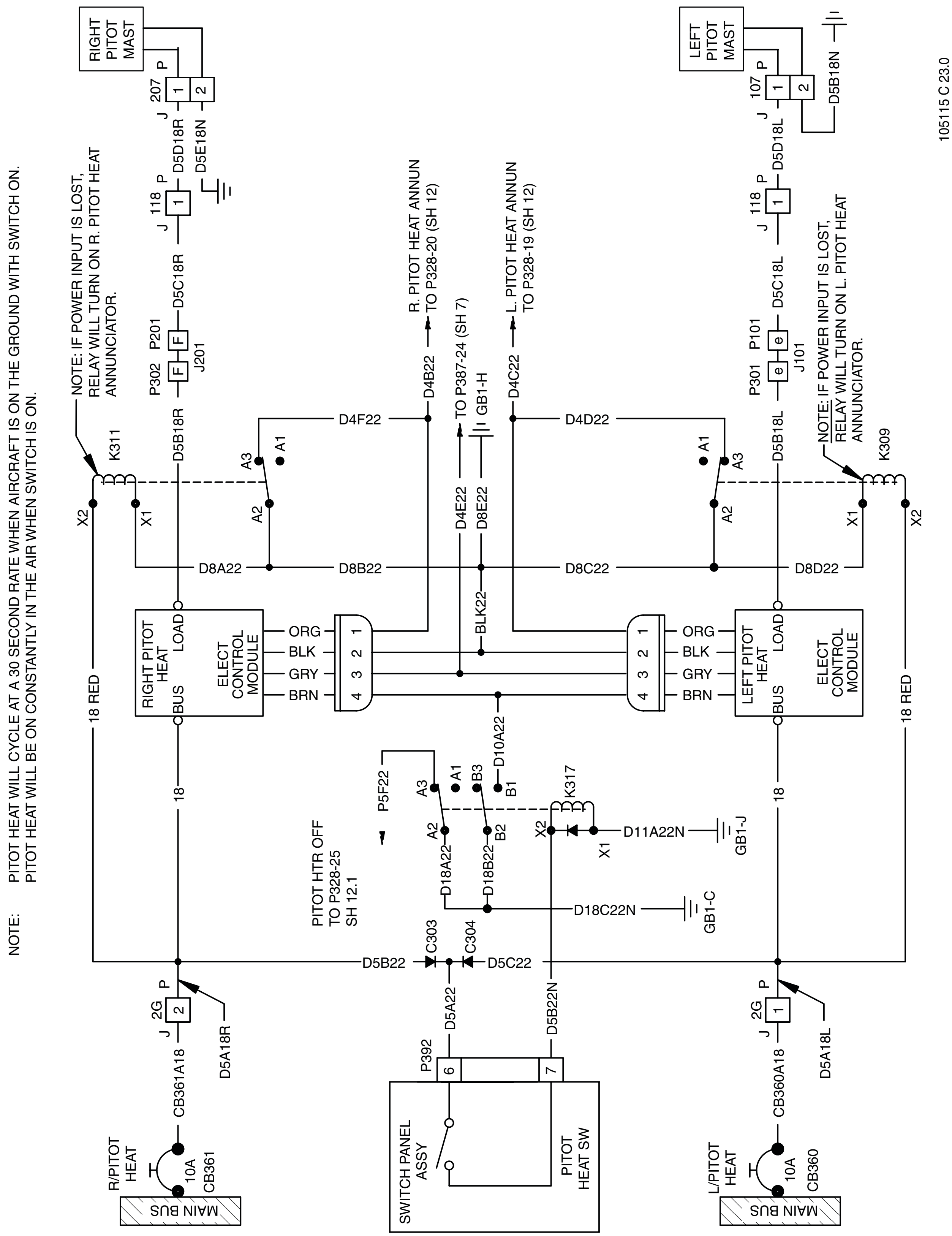
89801 AH 26.0



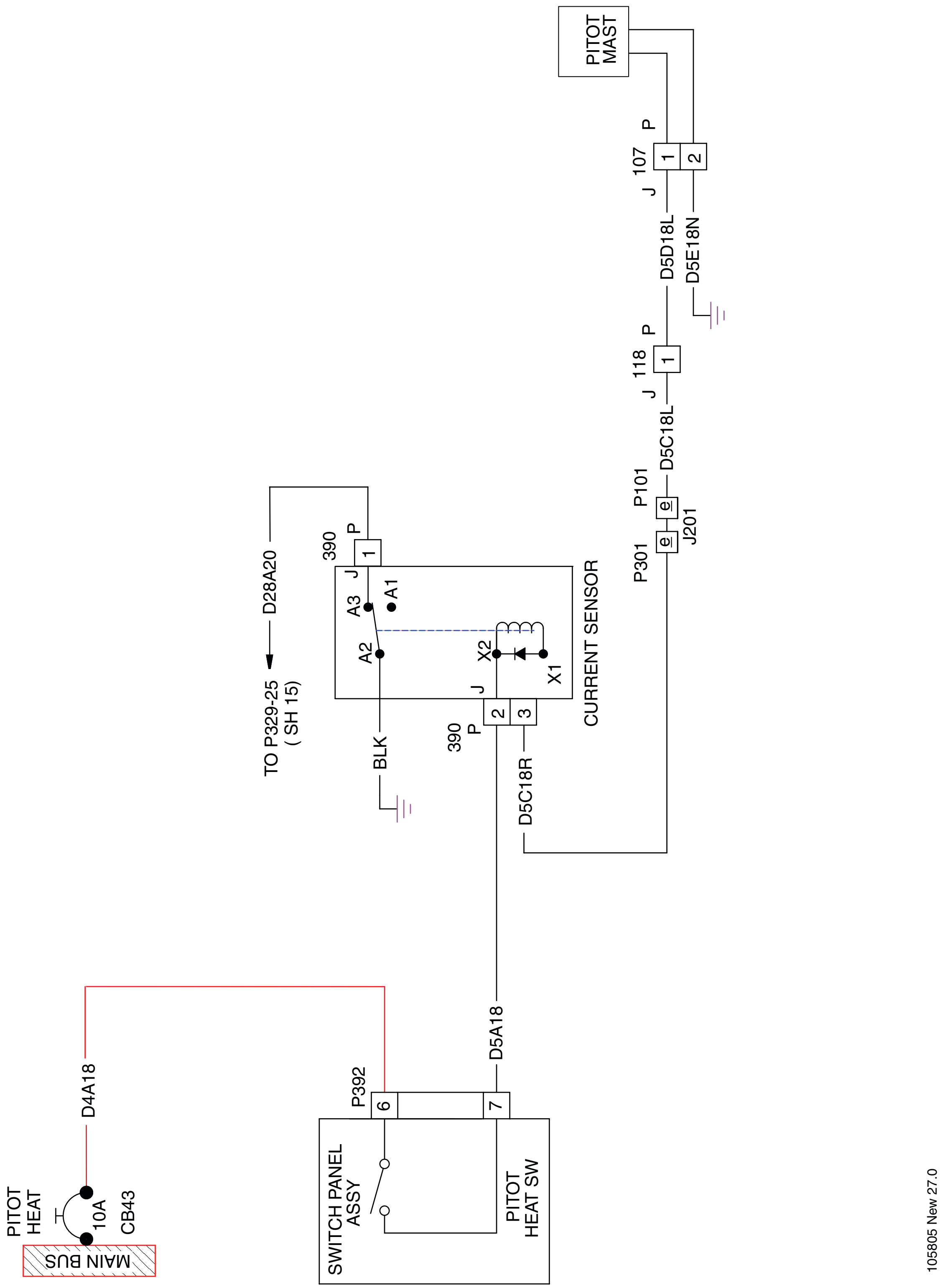
Pitot Heat
Figure 2 (Sheet 2 of 10)



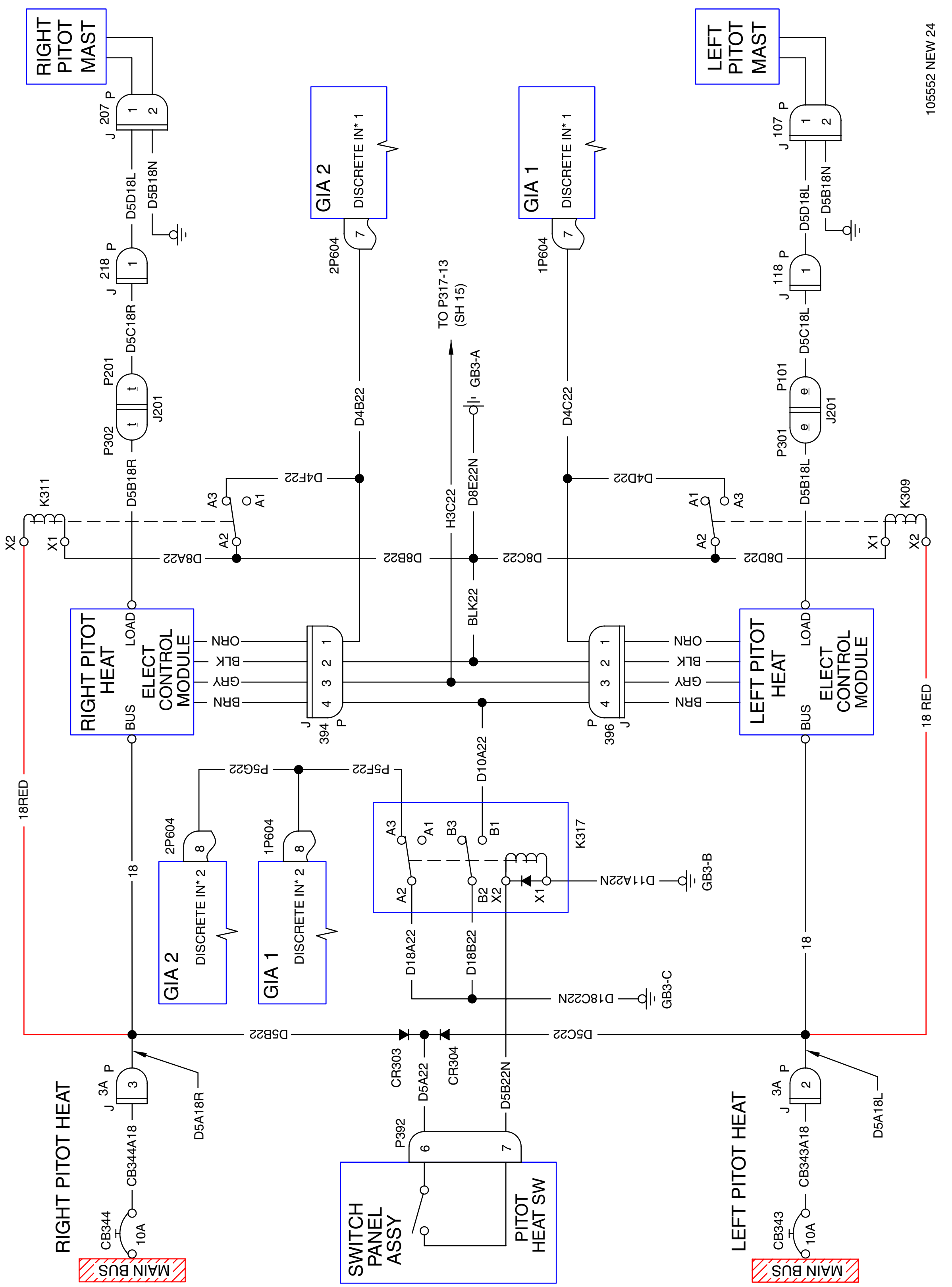
Pitot Heat
Figure 2 (Sheet 3 of 10)



Pitot Heat
Figure 2 (Sheet 4 of 10)

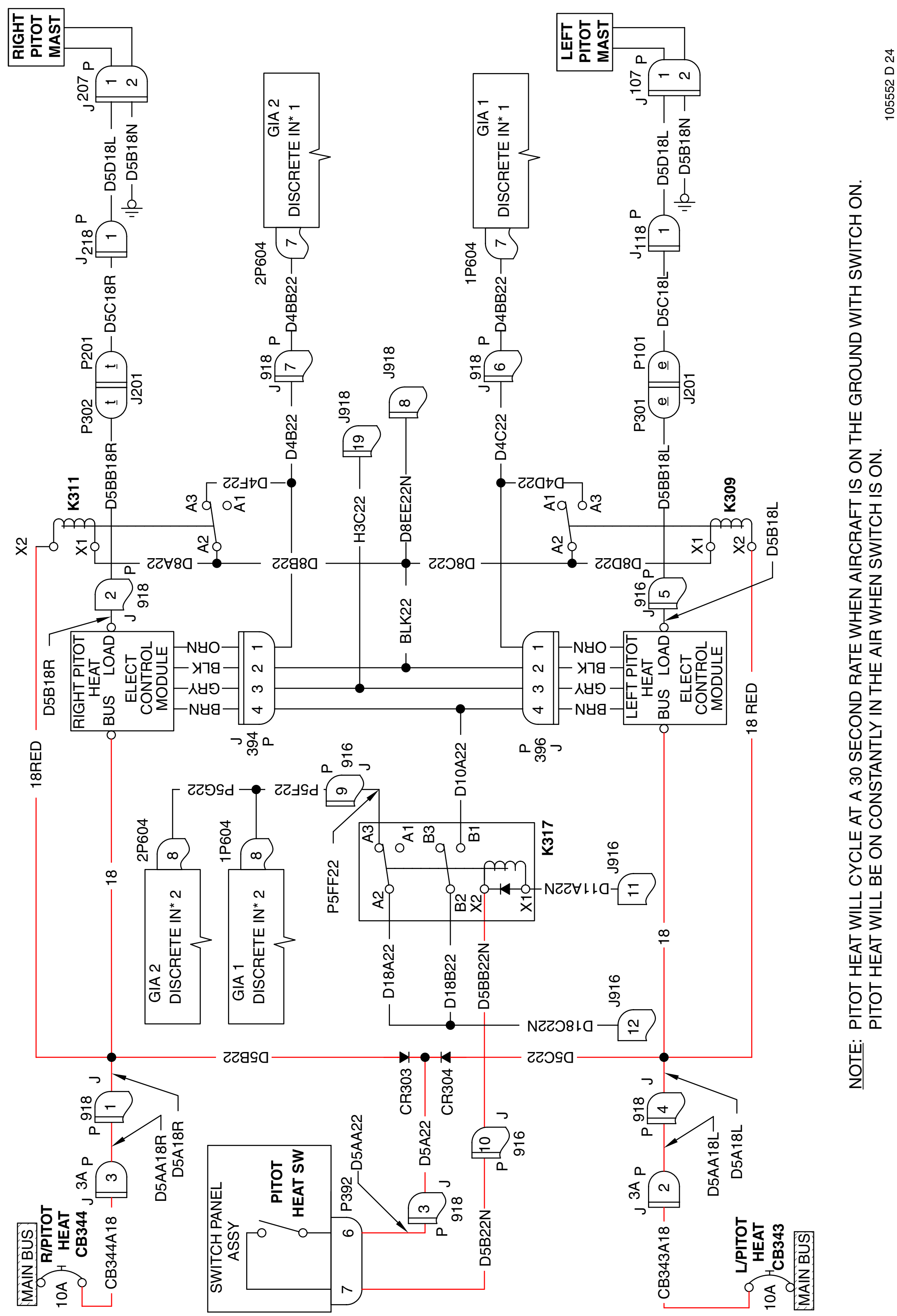


Pitot Heat
Figure 2 (Sheet 5 of 10)

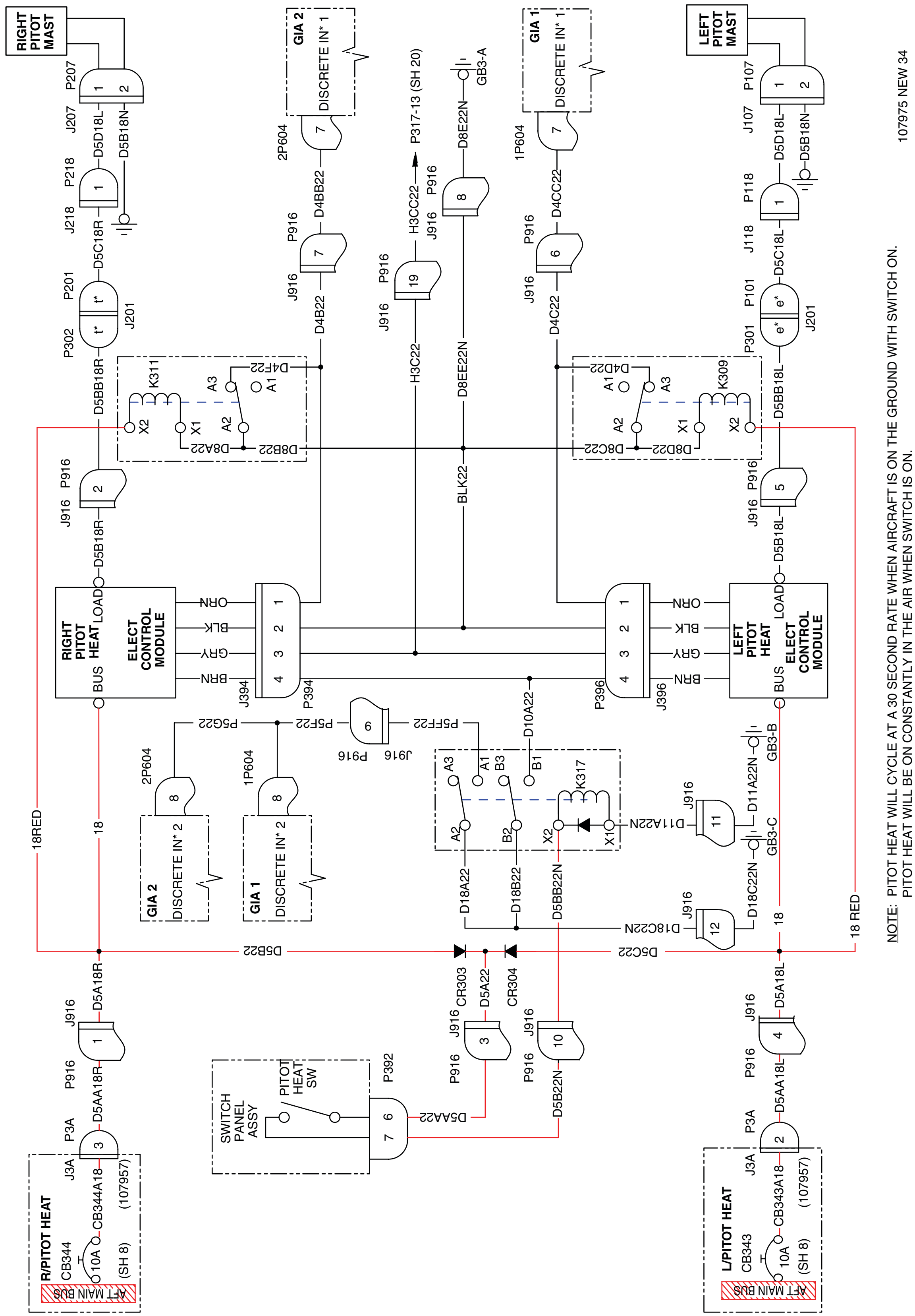


NOTE: Pitot heat will cycle at a 30-second rate when aircraft is on the ground with switch ON. Pitot heat will be ON constantly in the air when switch is ON.

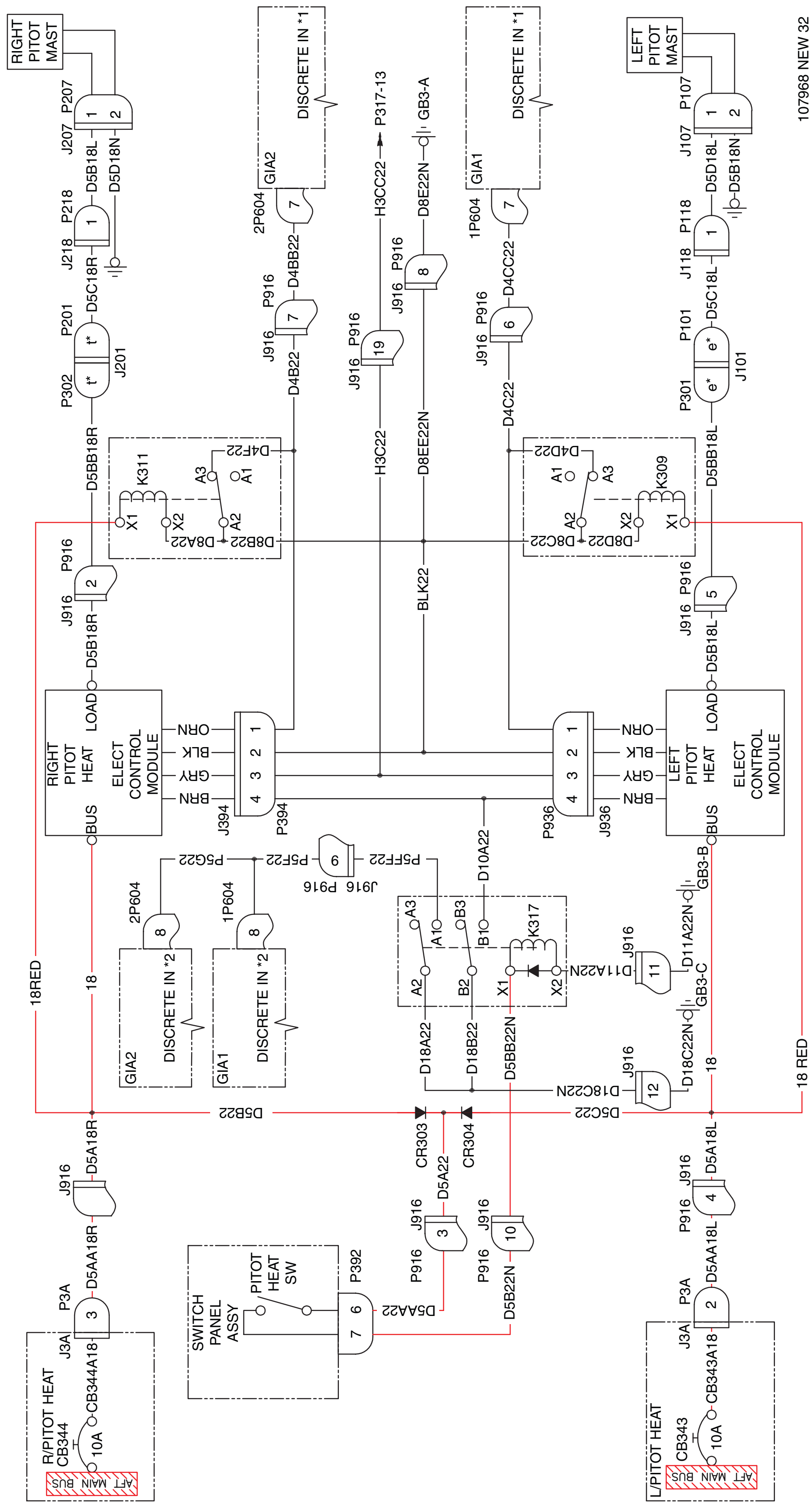
Pitot Heat
Figure 2 (Sheet 6 of 10)



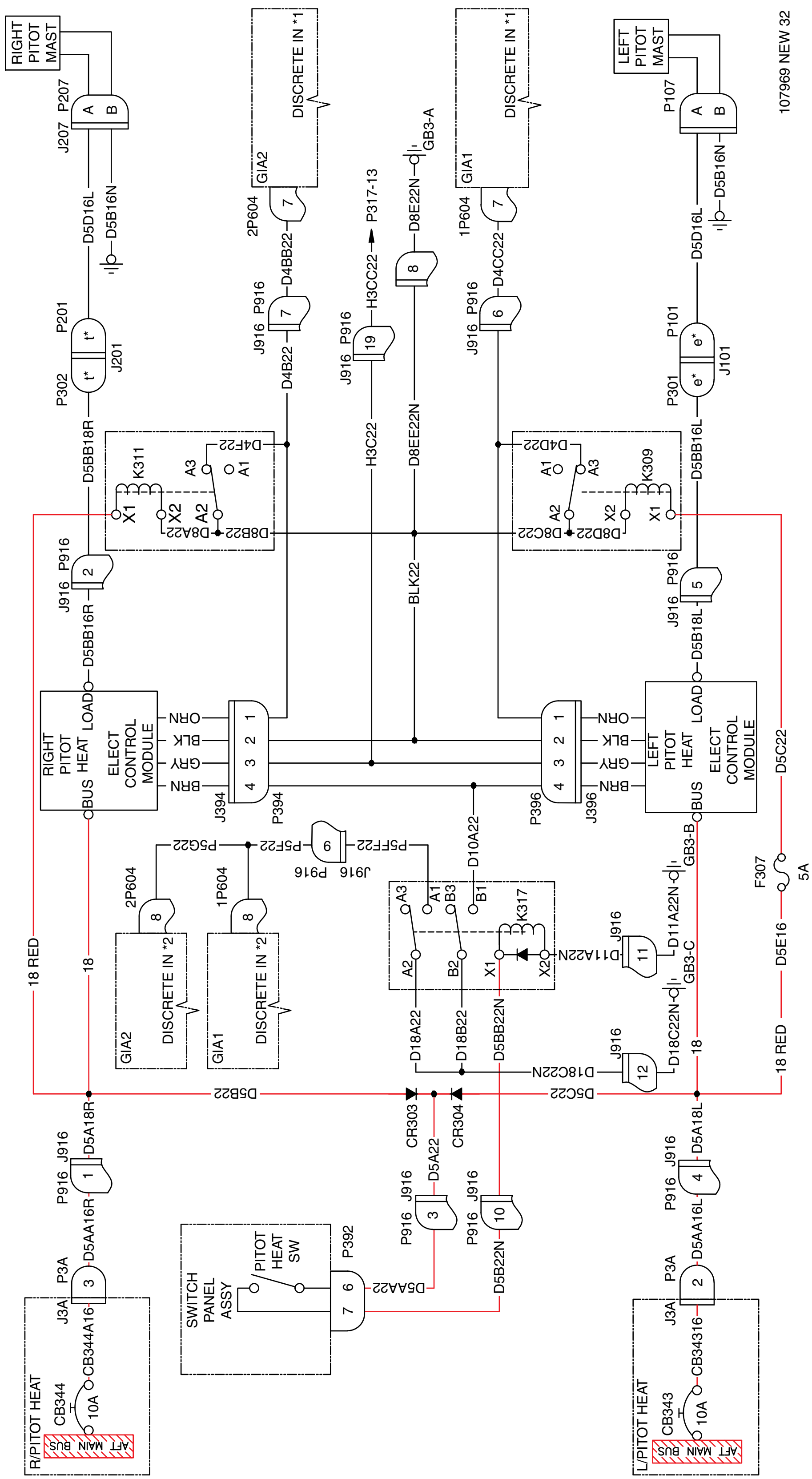
Pitot Heat
Figure 2 (Sheet 7 of 10)



Pitot Heat
Figure 2 (Sheet 8 of 10)

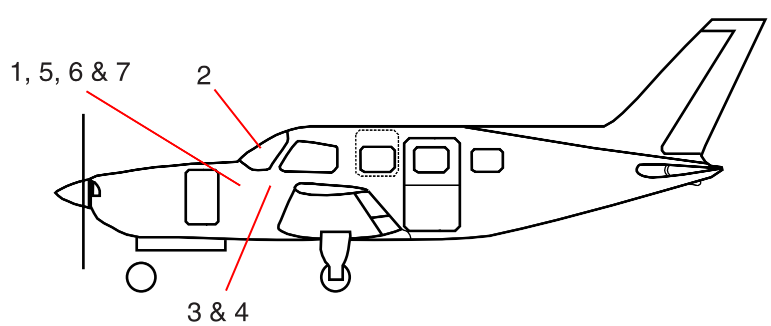


Pitot Heat
Figure 2 (Sheet 9 of 10)

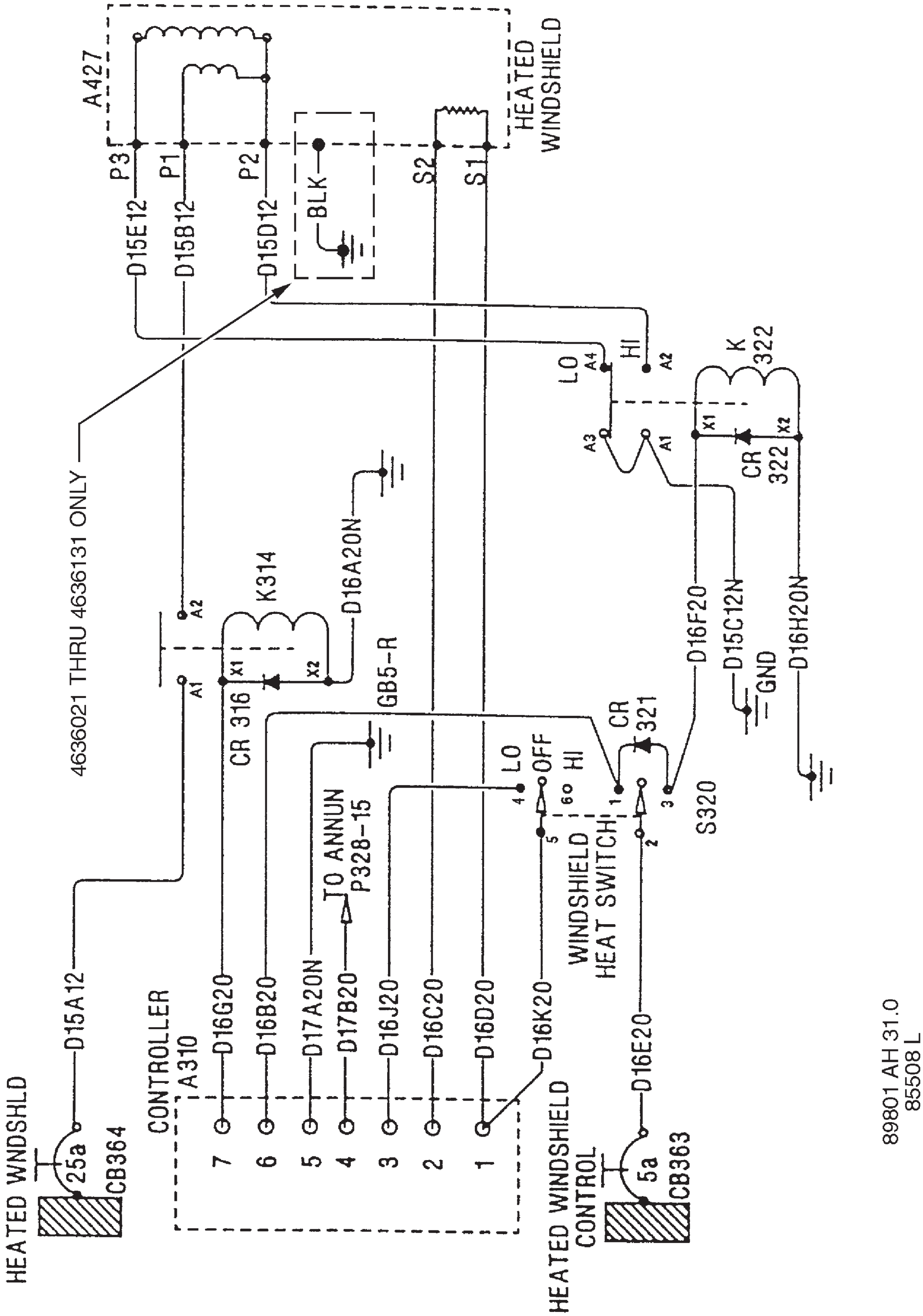


Pitot Heat
Figure 2 (Sheet 10 of 10)

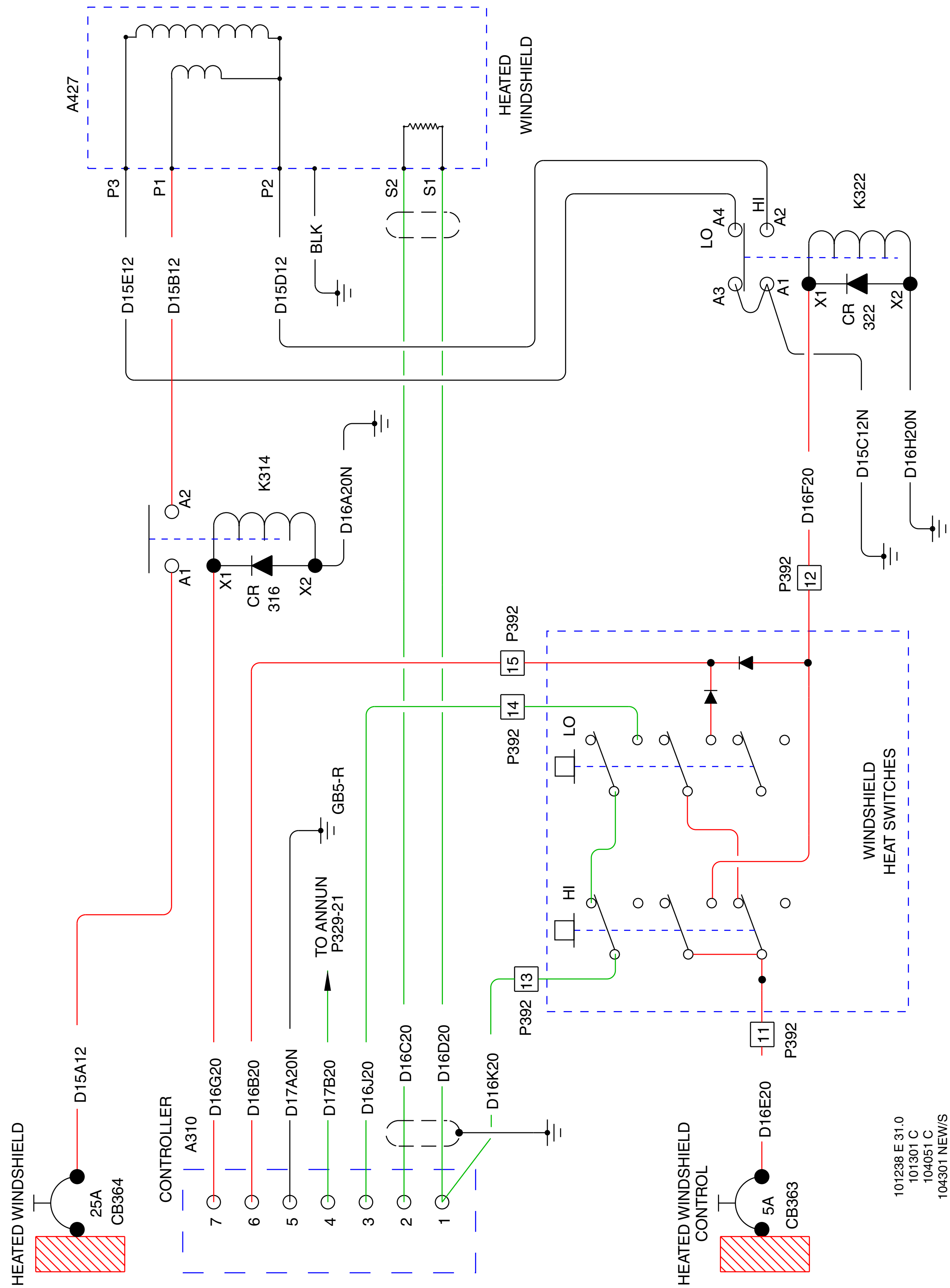
Item #	Designation	Description
1	A310	Controller
2	A427	Heated Windshield
3	CB363, CB46	Circuit Breaker - Heated Windshield Control (5 Amp)
4	CB364, CB47	Circuit Breaker - Heated Windshield (25 Amp)
5	K314	Controller Relay
6	K322	Heat Relay
7	S320	Heat Switch



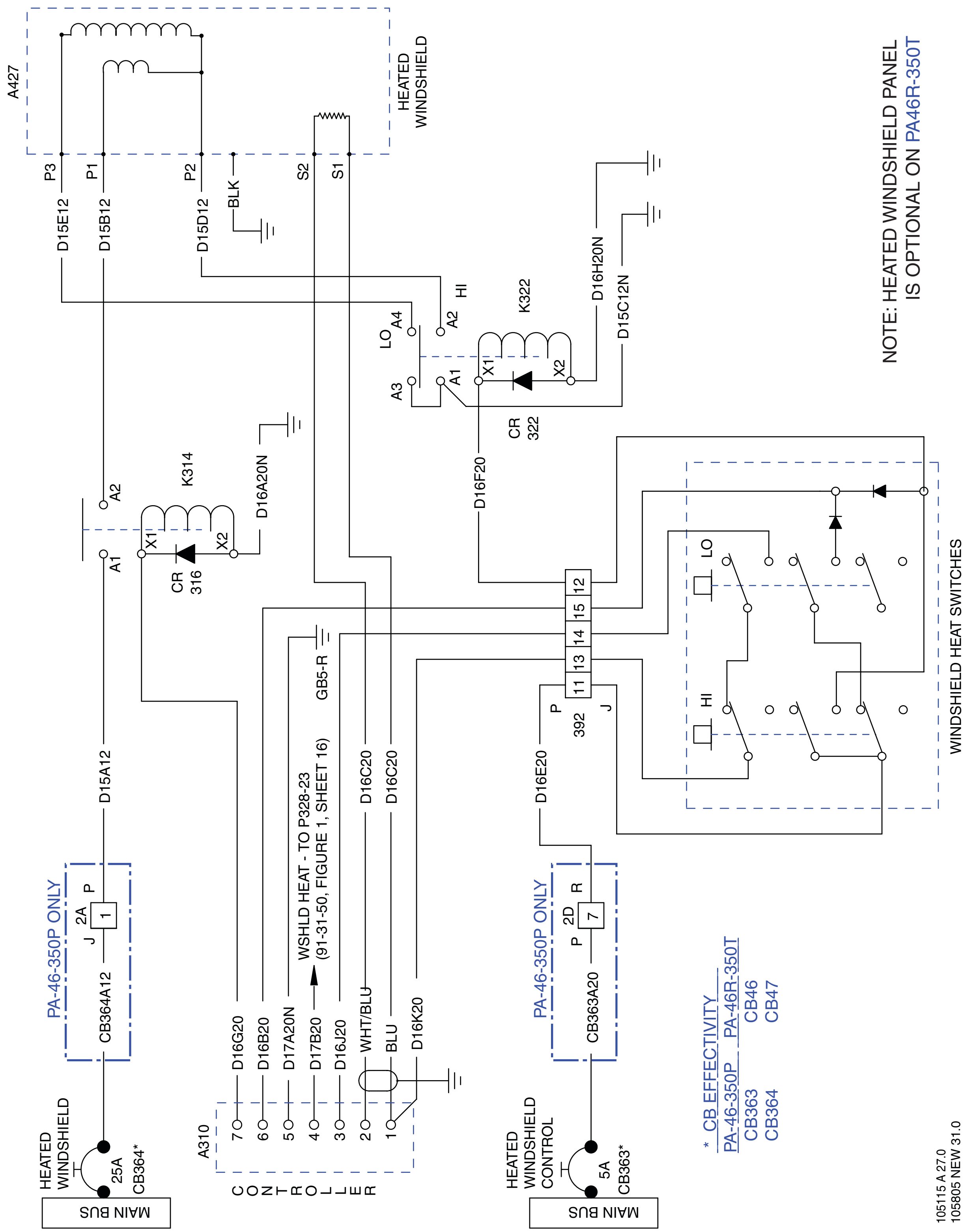
Heated Windshield (Optional)
Figure 1 (Sheet 1 of 5)



Heated Windshield (Optional)
Figure 1 (Sheet 2 of 5)

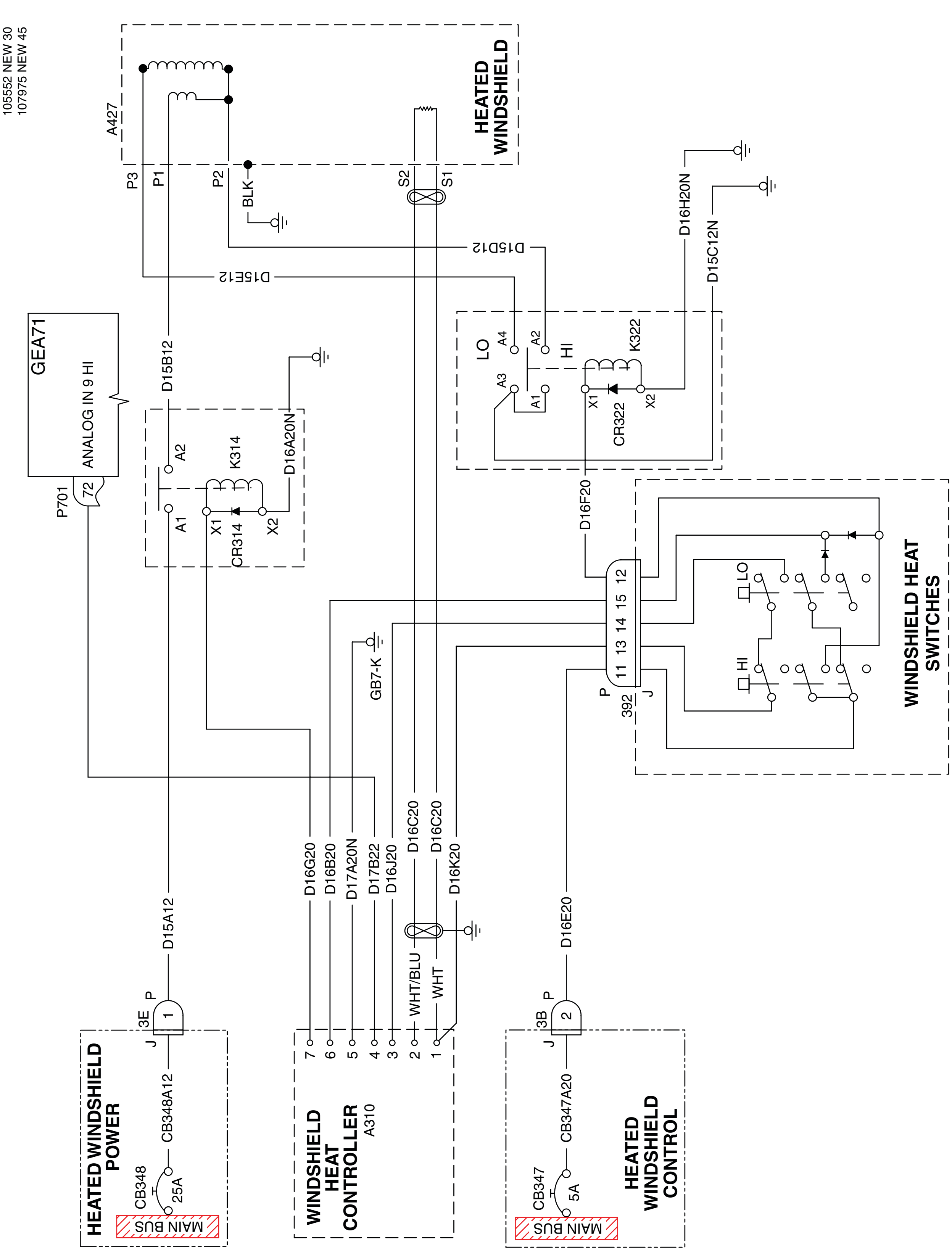


Heated Windshield (Optional)
Figure 1 (Sheet 3 of 5)



Heated Windshield (Optional)
Figure 1 (Sheet 4 of 5)

Effectivity
4636375–4636459, 4636461–4636462, 4636481
4692001–4692133, 4692141, 4692149, 4692153



Heated Windshield (Optional)
Figure 1 (Sheet 5 of 5)

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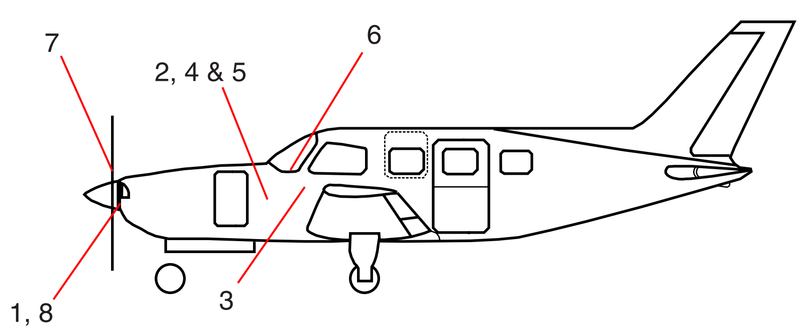
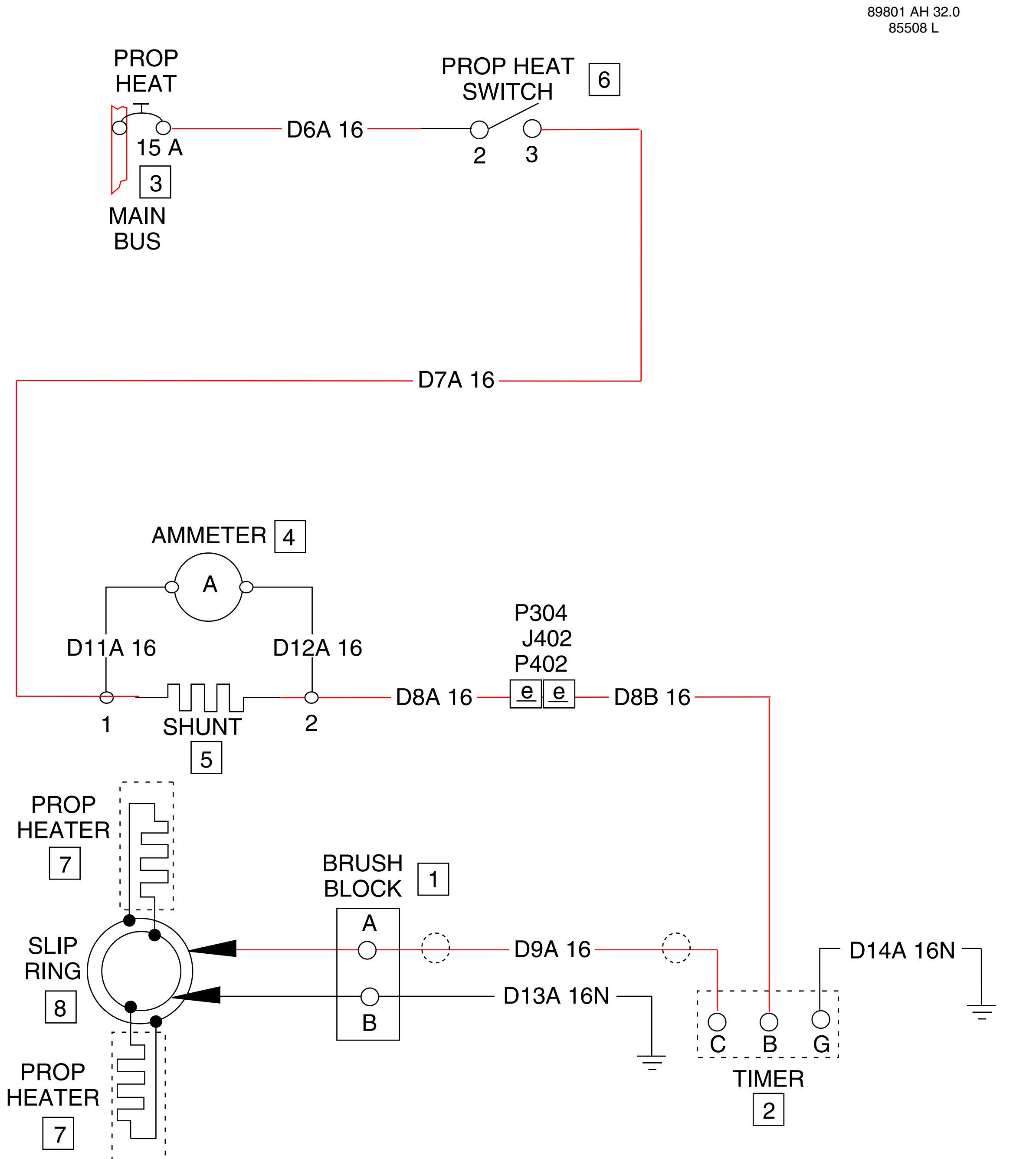
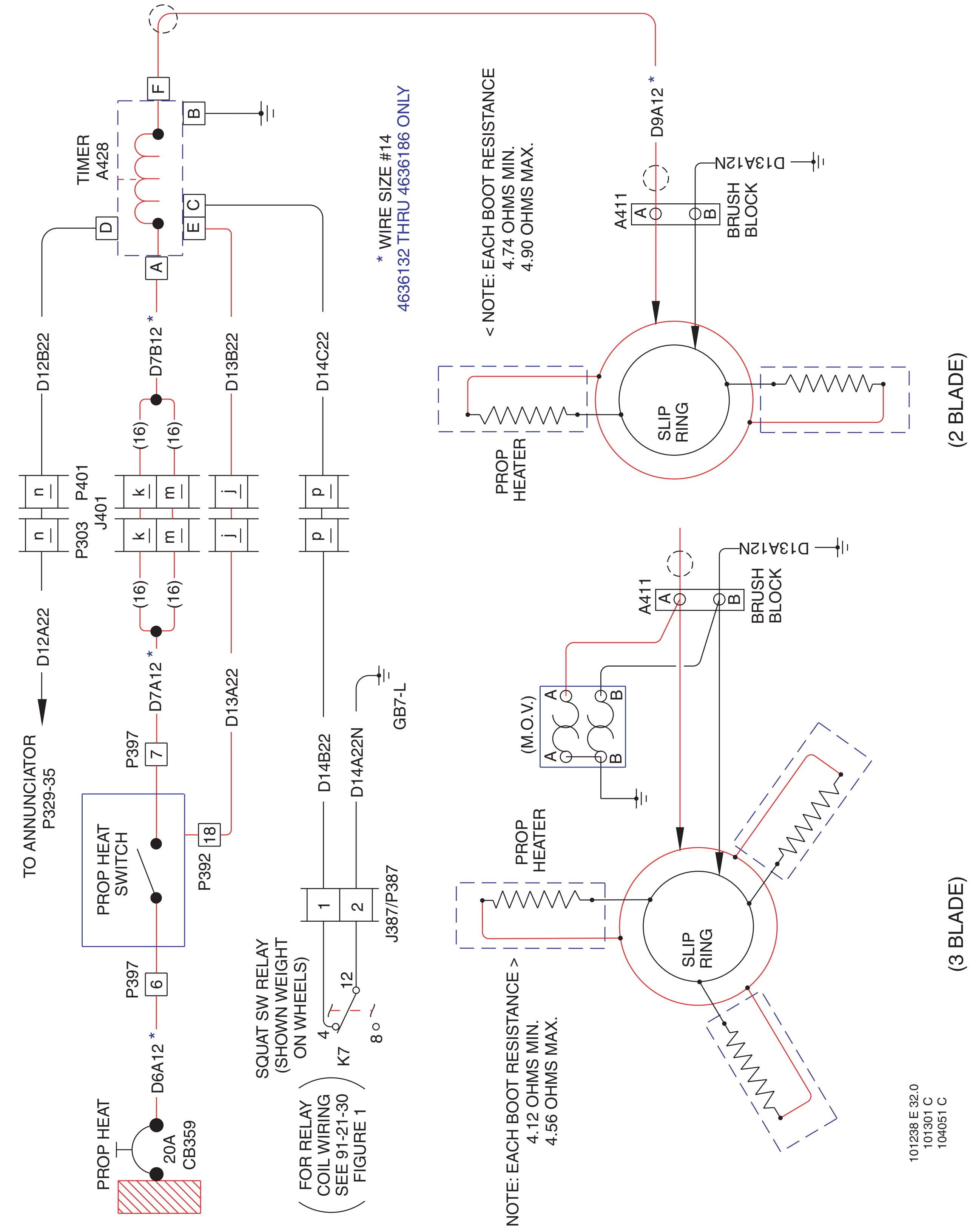
Item #	Designation	Description
1	A411	Brush Block
2	A428	Timer
3	CB358, CB359, CB44	Circuit Breaker - Prop Heat (15 or 20 Amp)
4	M303	Ammeter
5	R306	Shunt
6	S318	Switch
7		Prop Heater
8		Slip Ring



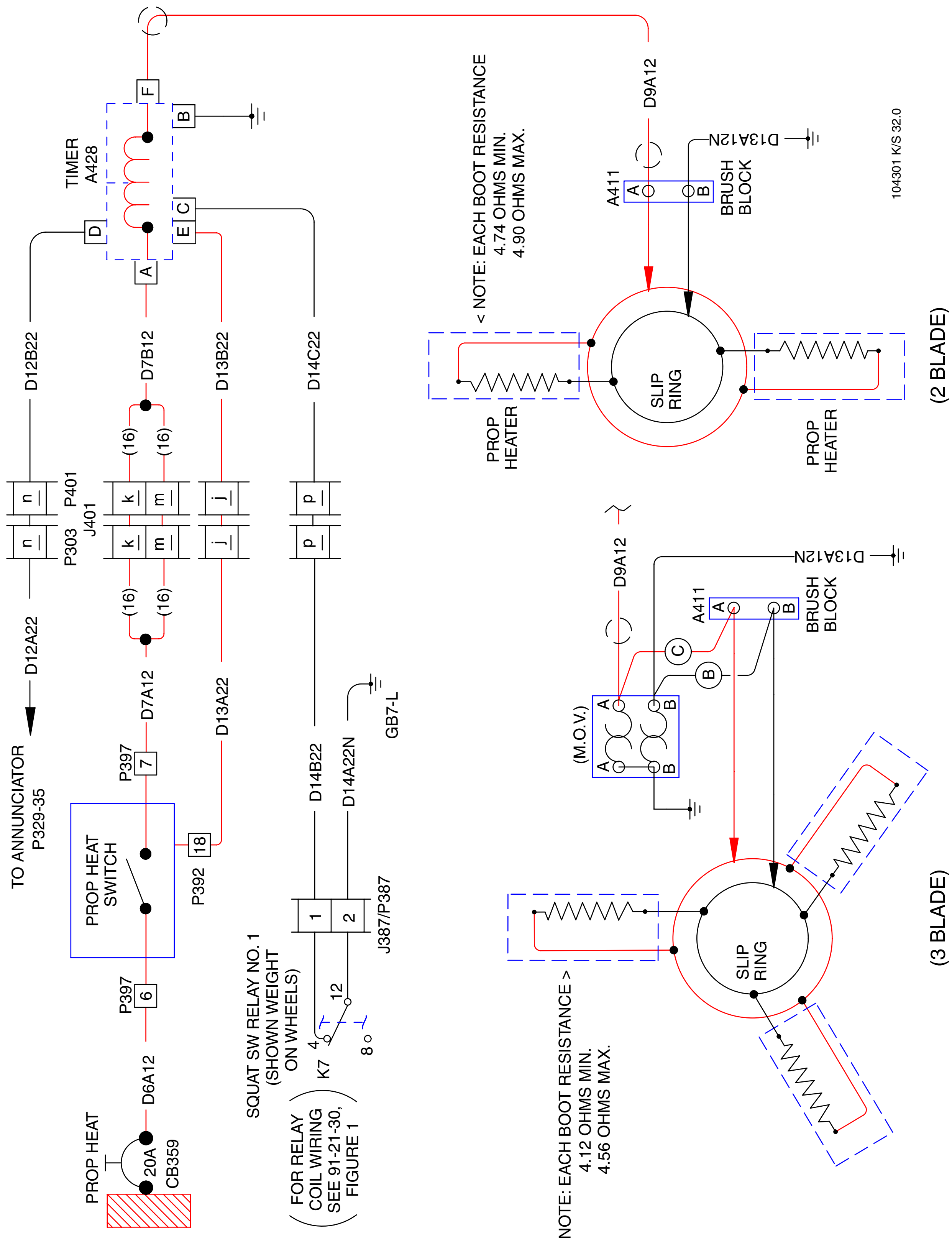
Propeller Heat (Optional)
Figure 1 (Sheet 1 of 7)



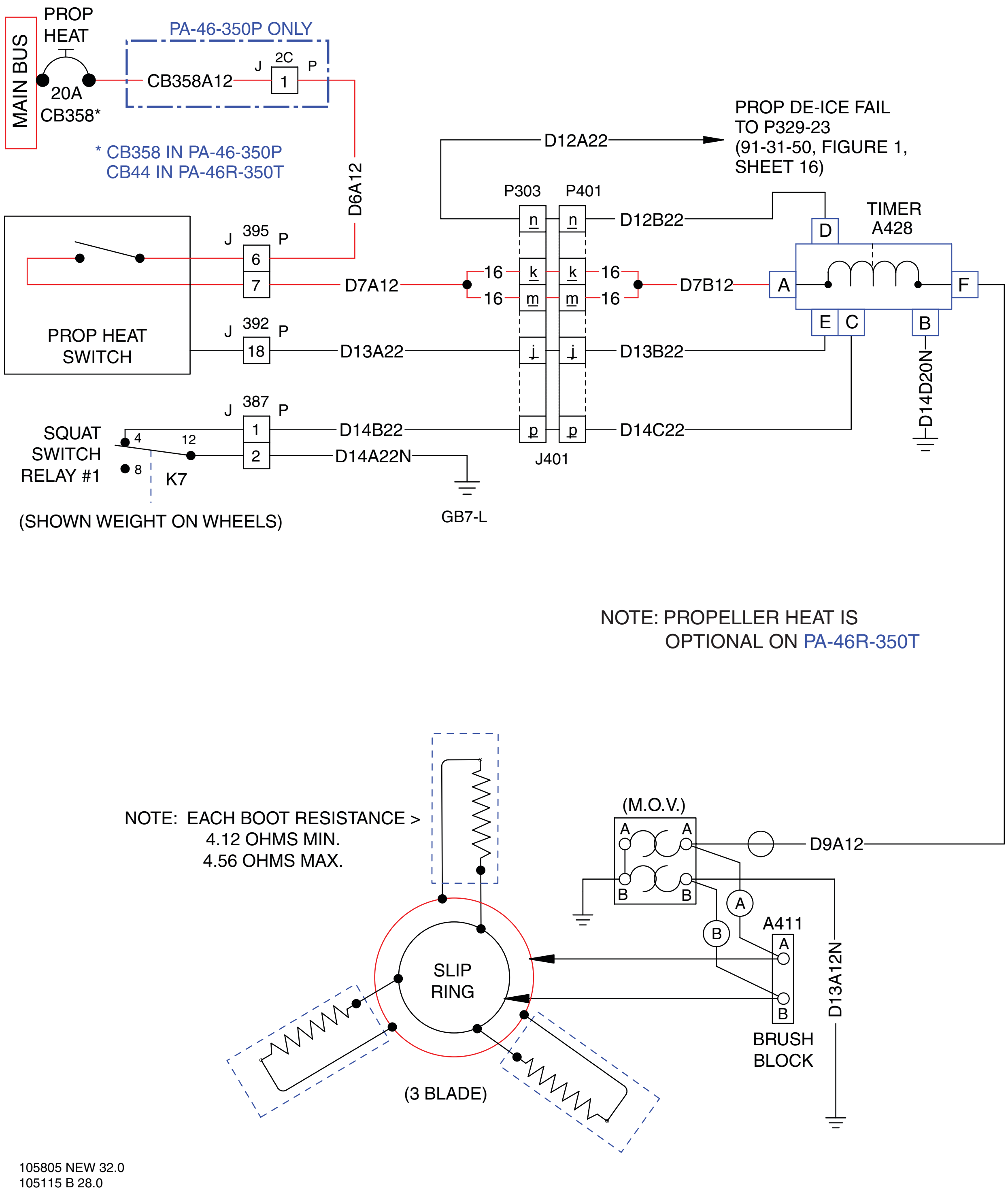
Propeller Heat (Optional)
Figure 1 (Sheet 2 of 7)



Propeller Heat (Optional)
Figure 1 (Sheet 3 of 7)

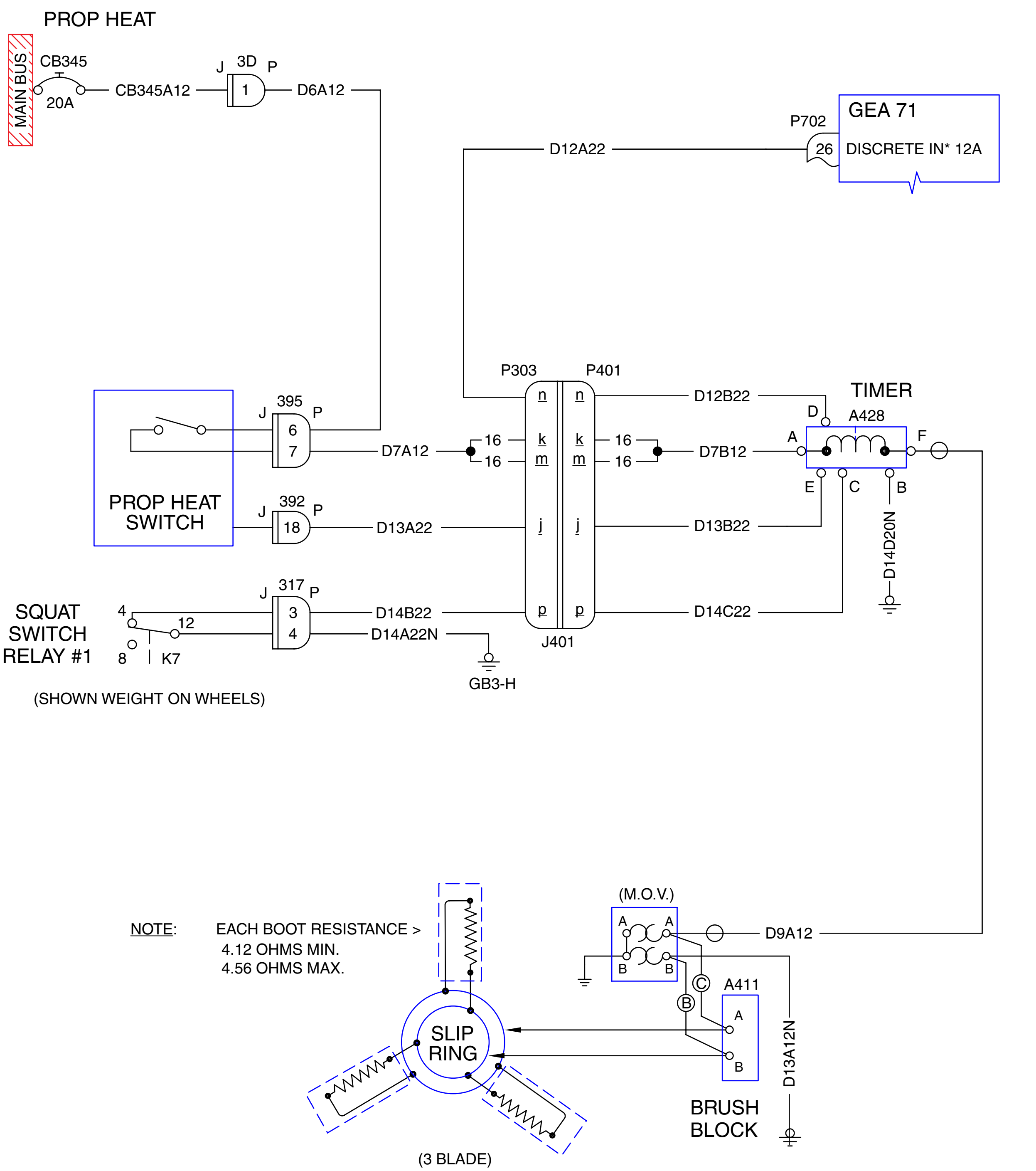


Propeller Heat (Optional)
Figure 1 (Sheet 4 of 7)

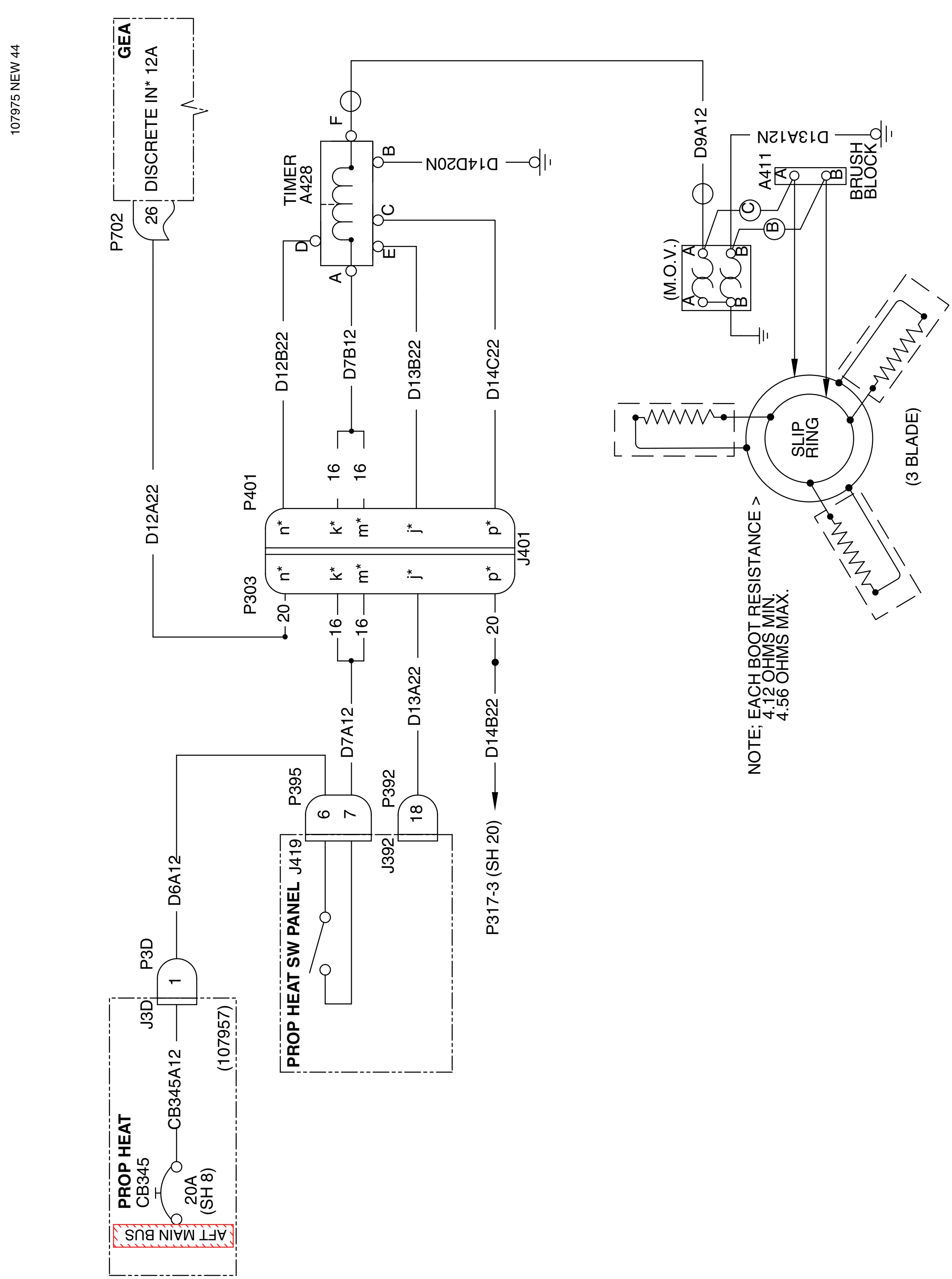


Propeller Heat (Optional)
Figure 1 (Sheet 5 of 7)

105552 NEW 28



Propeller Heat (Optional)
Figure 1 (Sheet 6 of 7)



Propeller Heat (Optional)
Figure 1 (Sheet 7 of 7)

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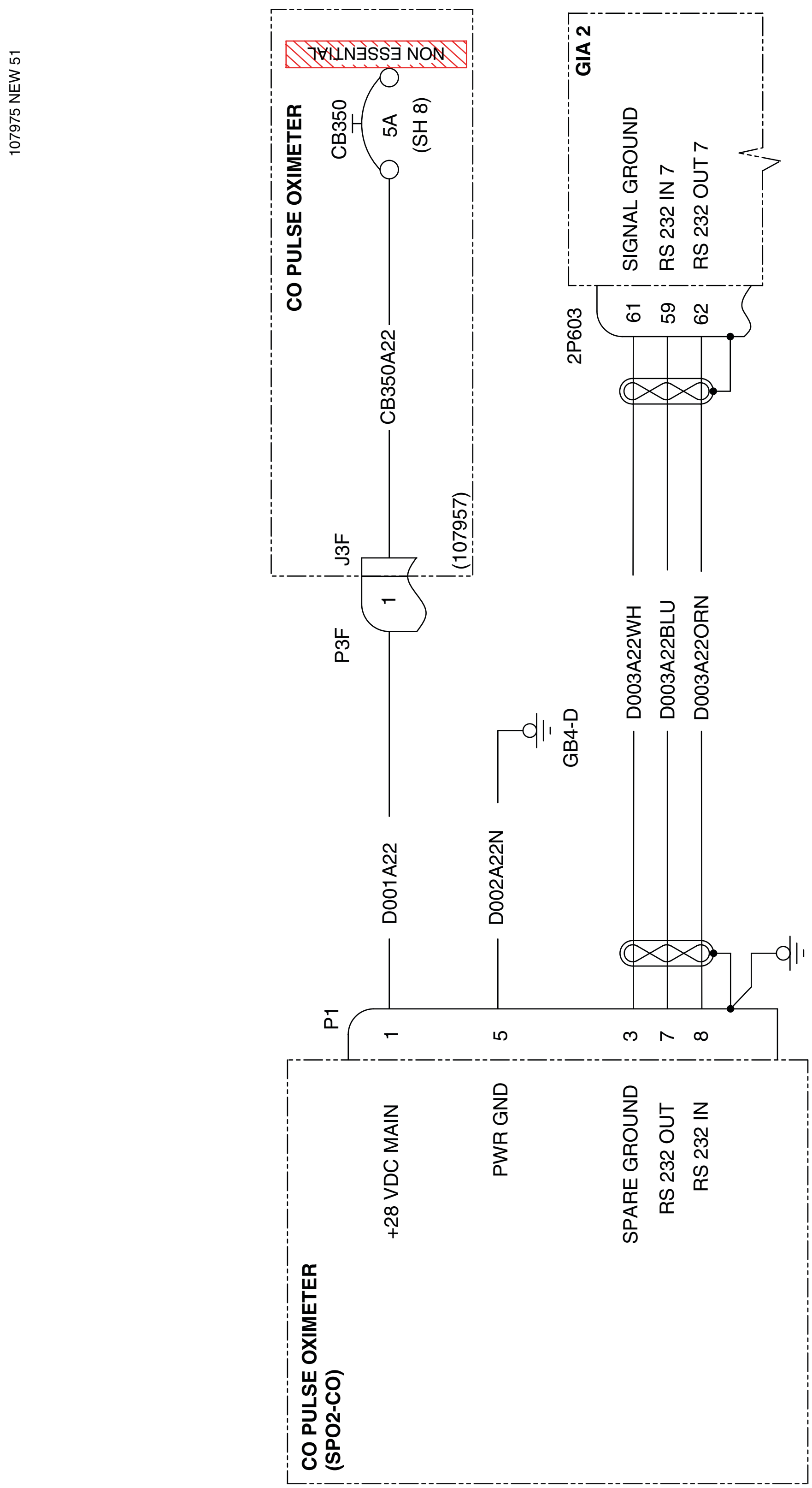

CO Pulse Oximeter
Figure 1

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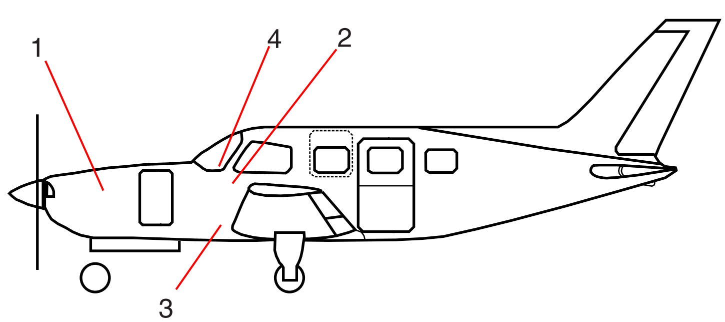
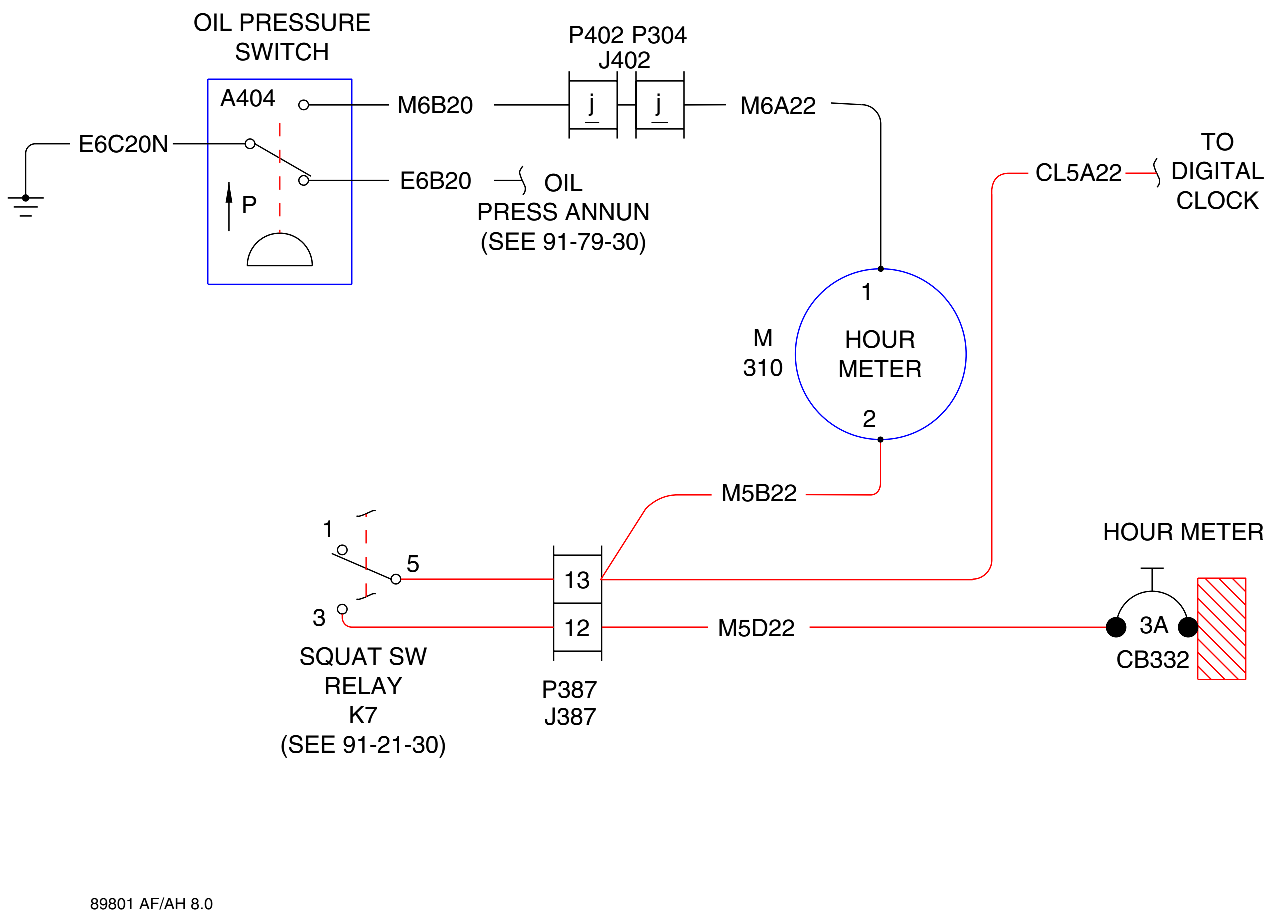
Item #	Designation	Description
1	A404	Oil Pressure Switch
2	CB332, CB40	Circuit Breaker - Hour Meter (3 Amp)
3	K7	Squat Switch Relay
4	M310	Flight Hour Meter

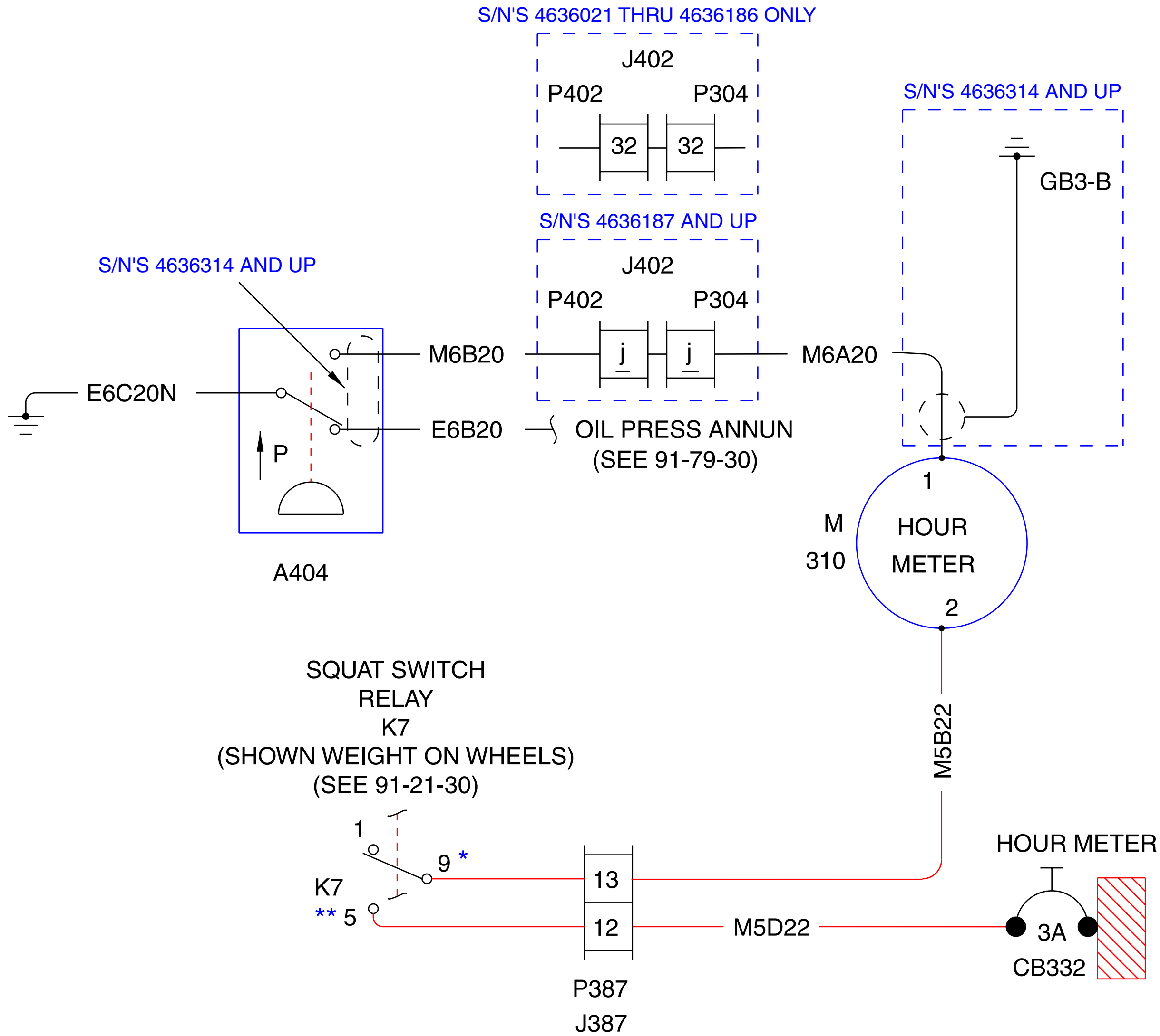
In S/N'S 4636375–4636459, 4636461–4636462, 4636481, see 105115 SH 7.



Flight Hour Meter
Figure 1 (Sheet 1 of 6)



Flight Hour Meter
Figure 1 (Sheet 2 of 6)

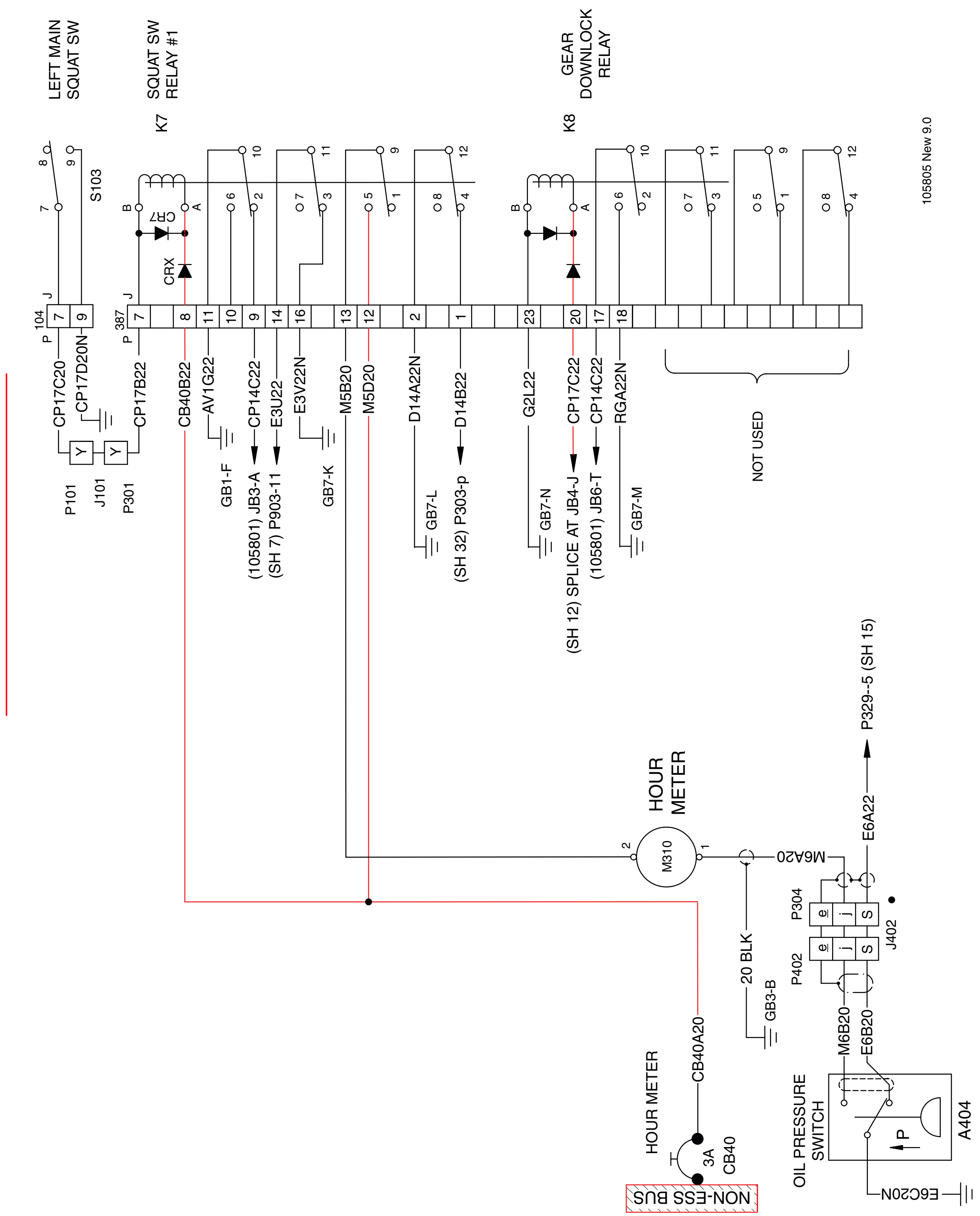


* 5 IN S/N'S 4636021 THRU 4636131 ONLY
** 3 IN S/N'S 4636021 THRU 4636131 ONLY

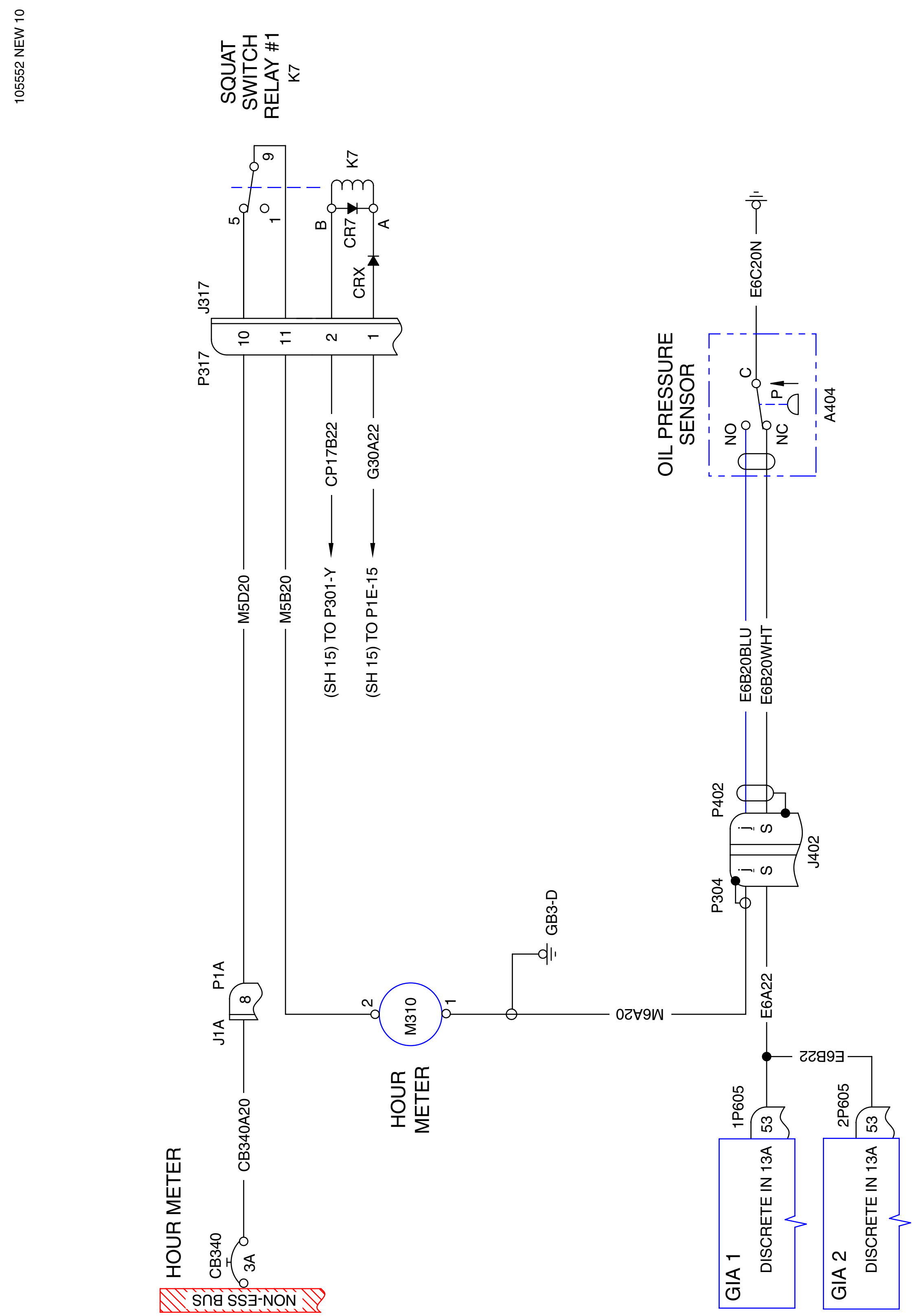
85508 L 8.0
101238 E
101301 C
104051 C
104301 L/S

IN S/N'S 4636375 AND UP, SEE 105115 Sheet 7.

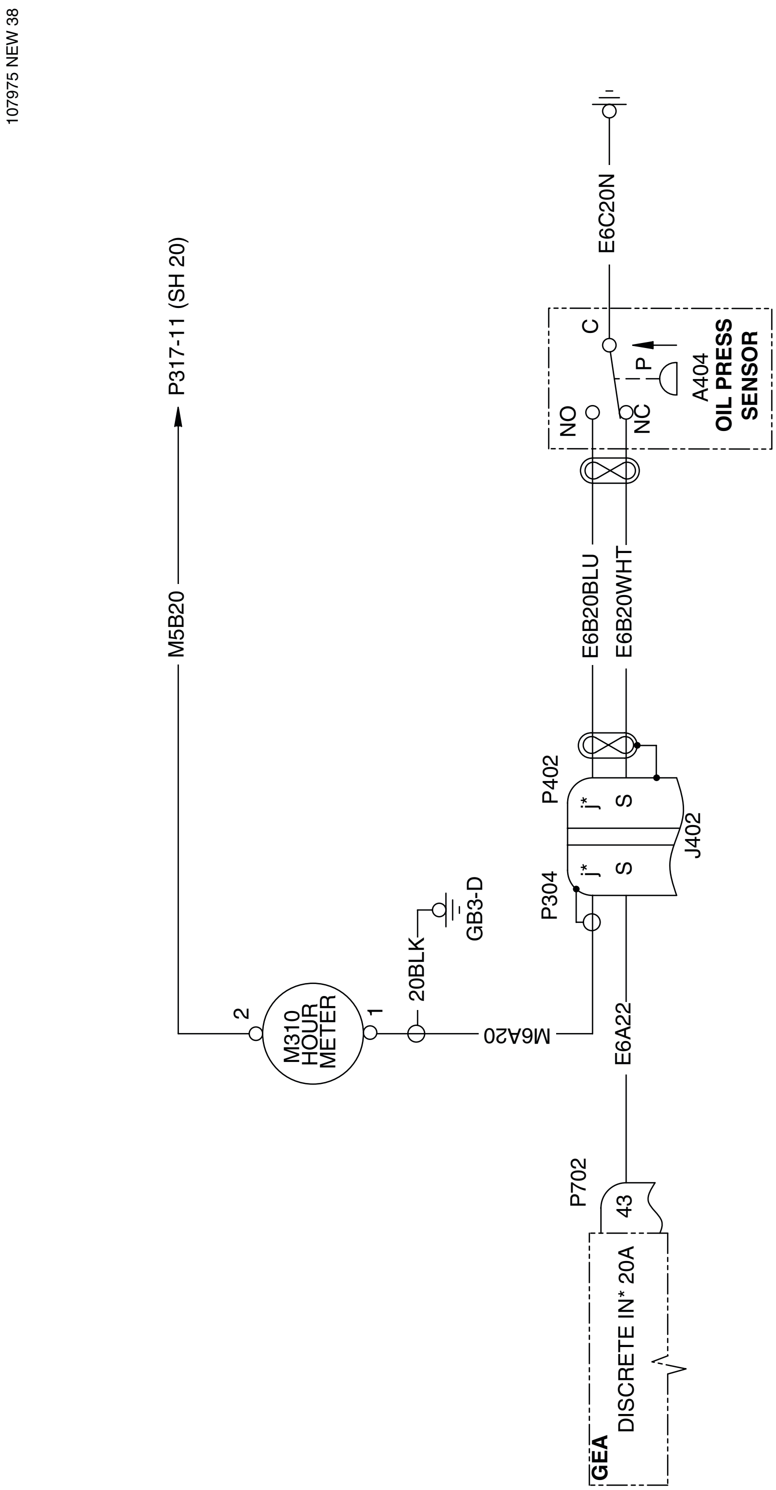
Flight Hour Meter
Figure 1 (Sheet 3 of 6)



Flight Hour Meter
Figure 1 (Sheet 4 of 6)

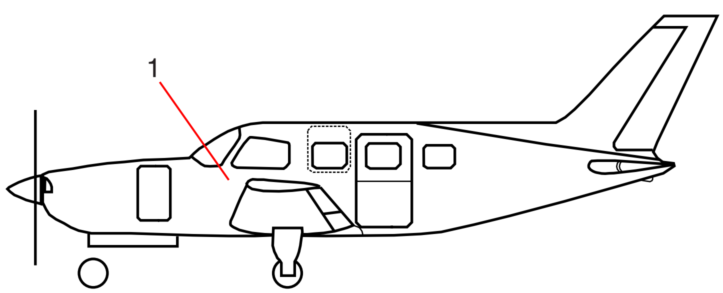


Flight Hour Meter
Figure 1 (Sheet 5 of 6)

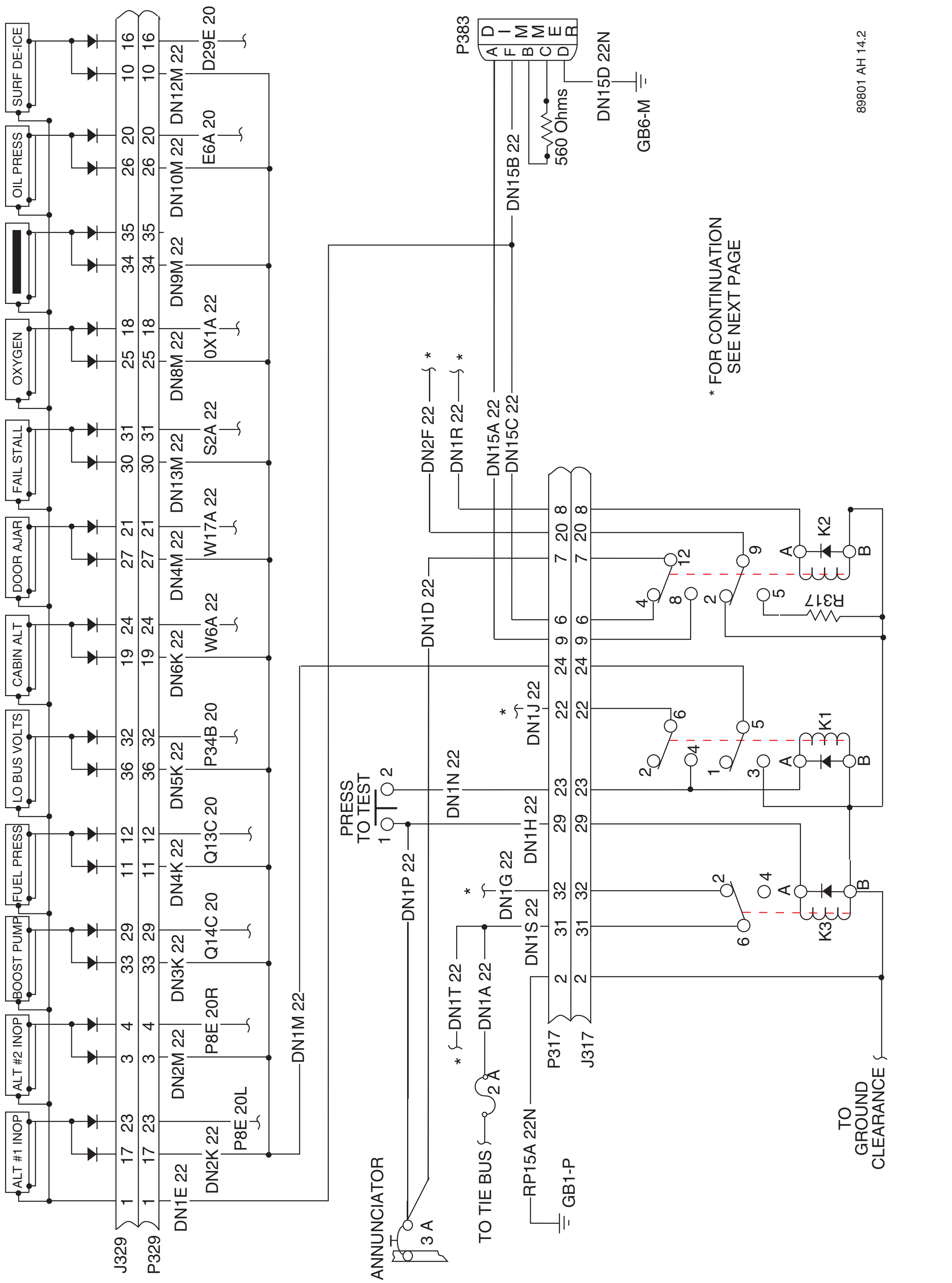


Flight Hour Meter
Figure 1 (Sheet 6 of 6)

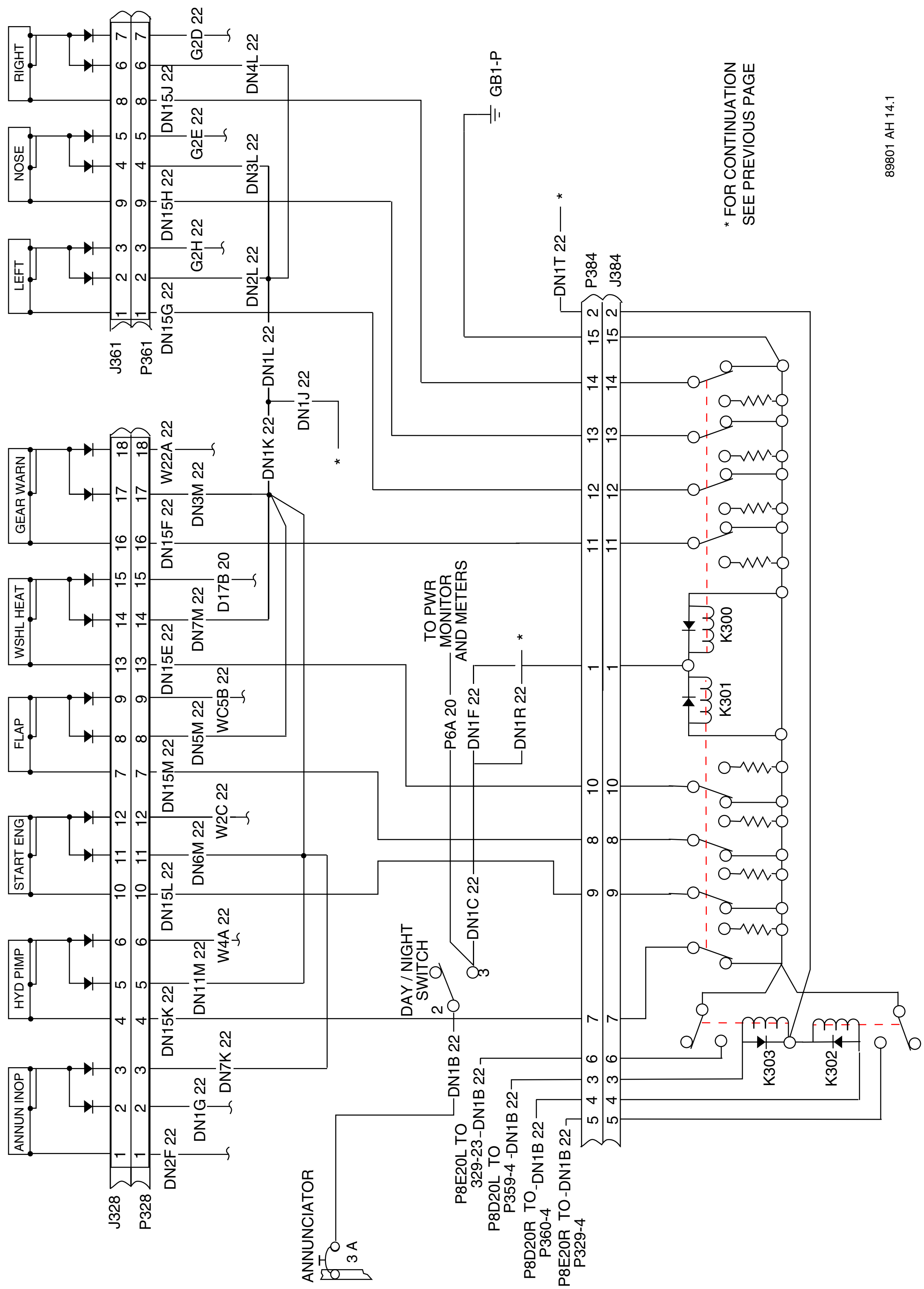
Item #	Designation	Description
1	CB355, CB6	Circuit Breaker - Annunciator (3 Amp)



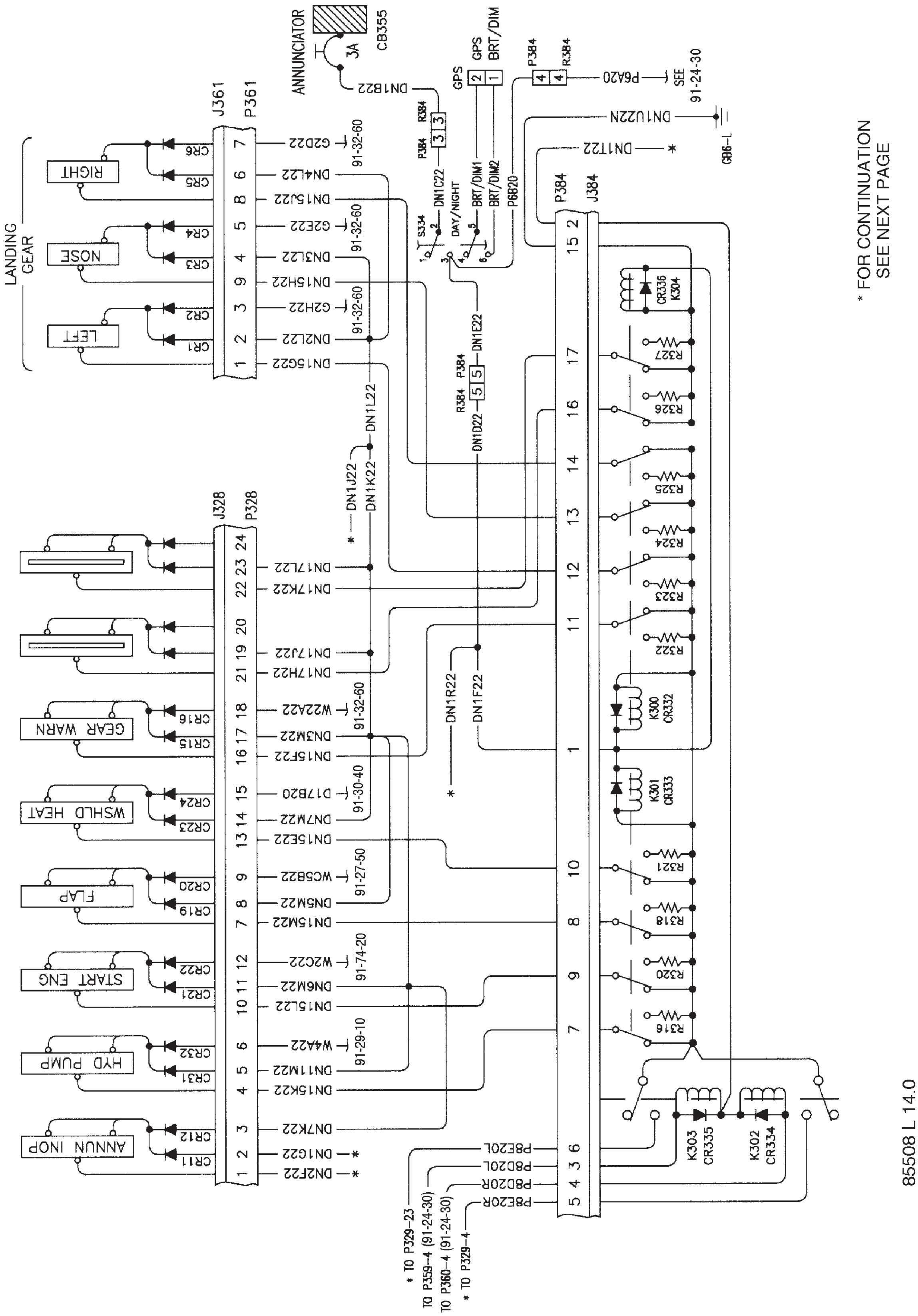
Annunciator Panel
Figure 1 (Sheet 1 of 18)



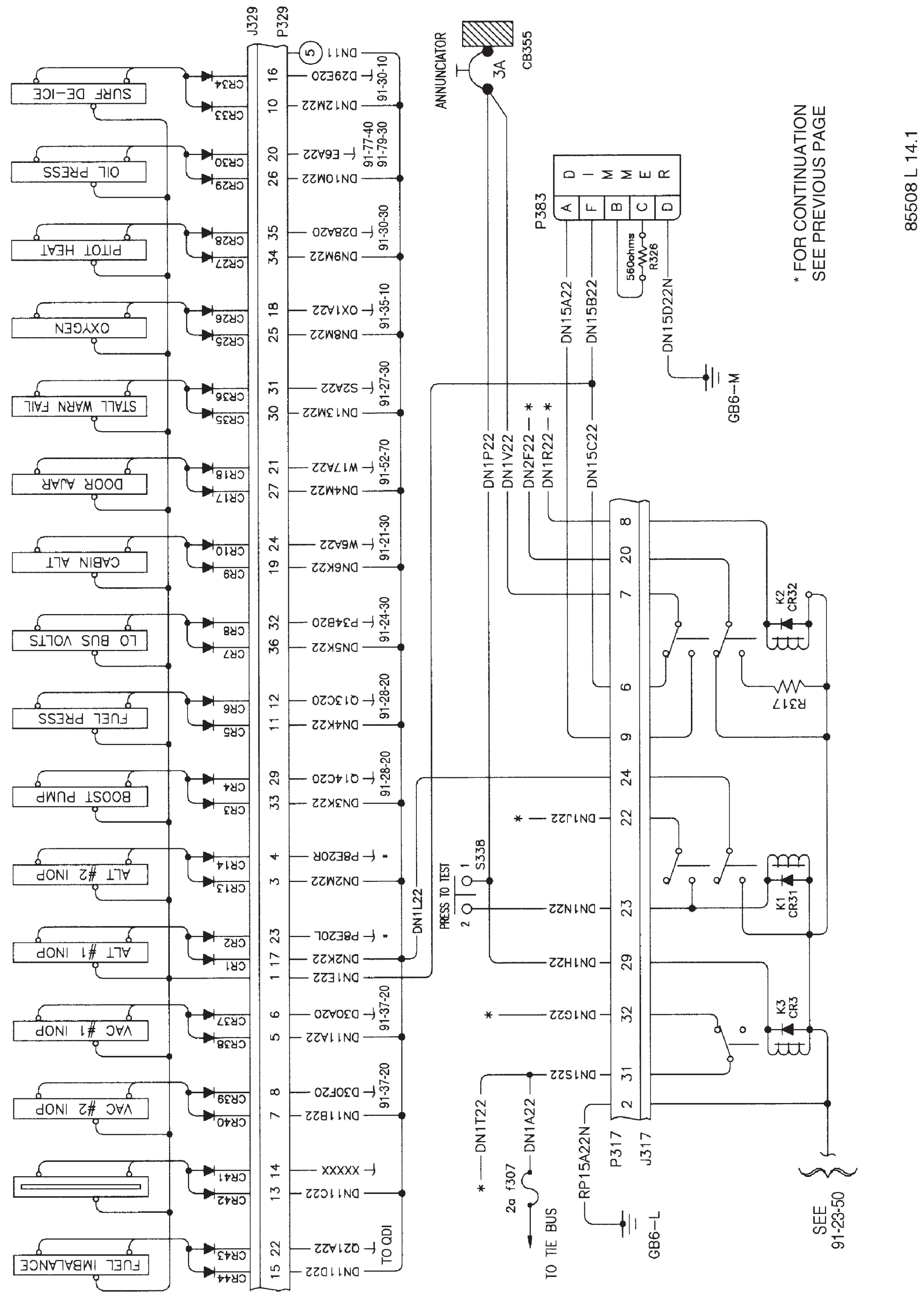
Annunciator Panel
Figure 1 (Sheet 2 of 18)



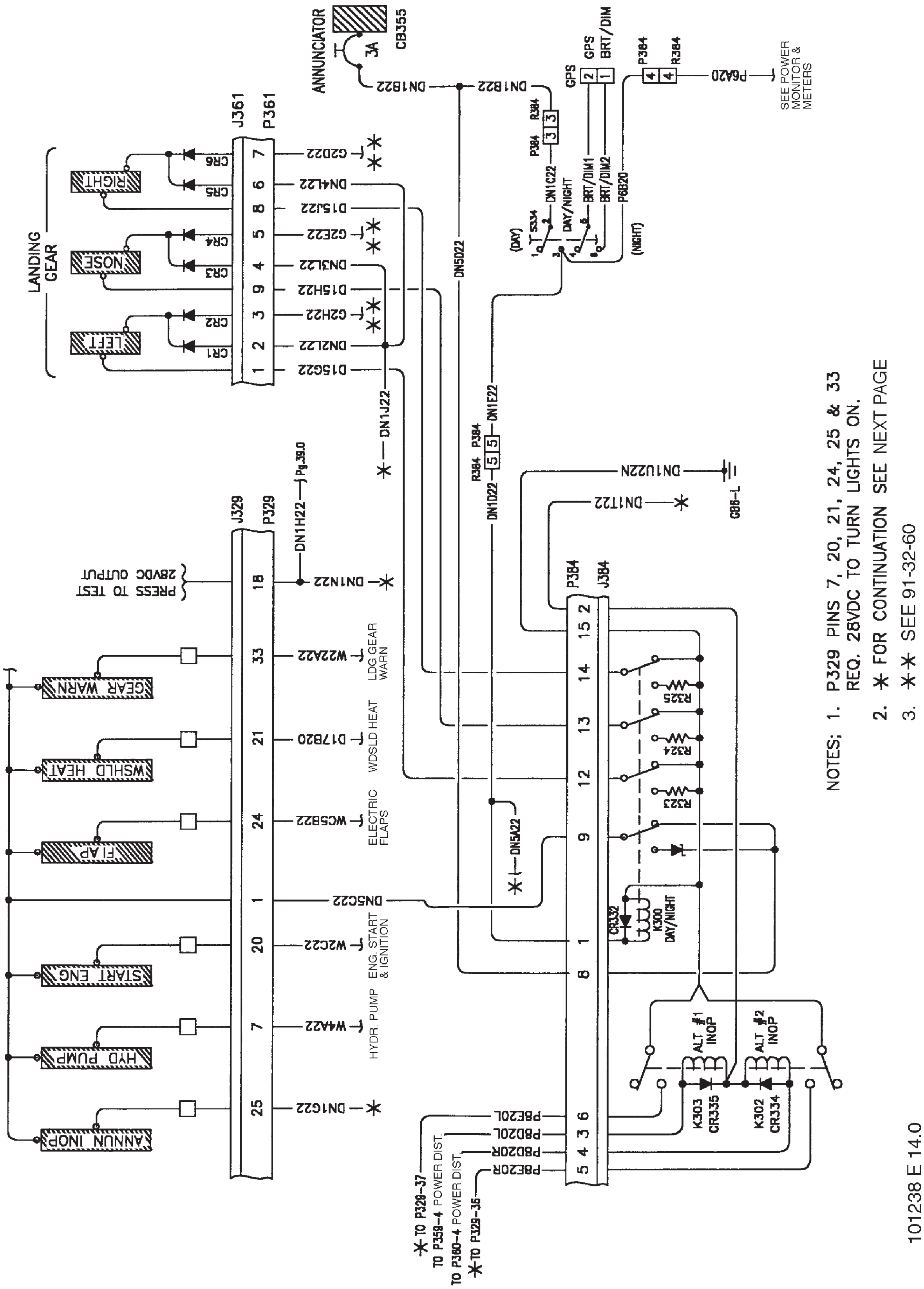
Annunciator Panel
Figure 1 (Sheet 3 of 18)



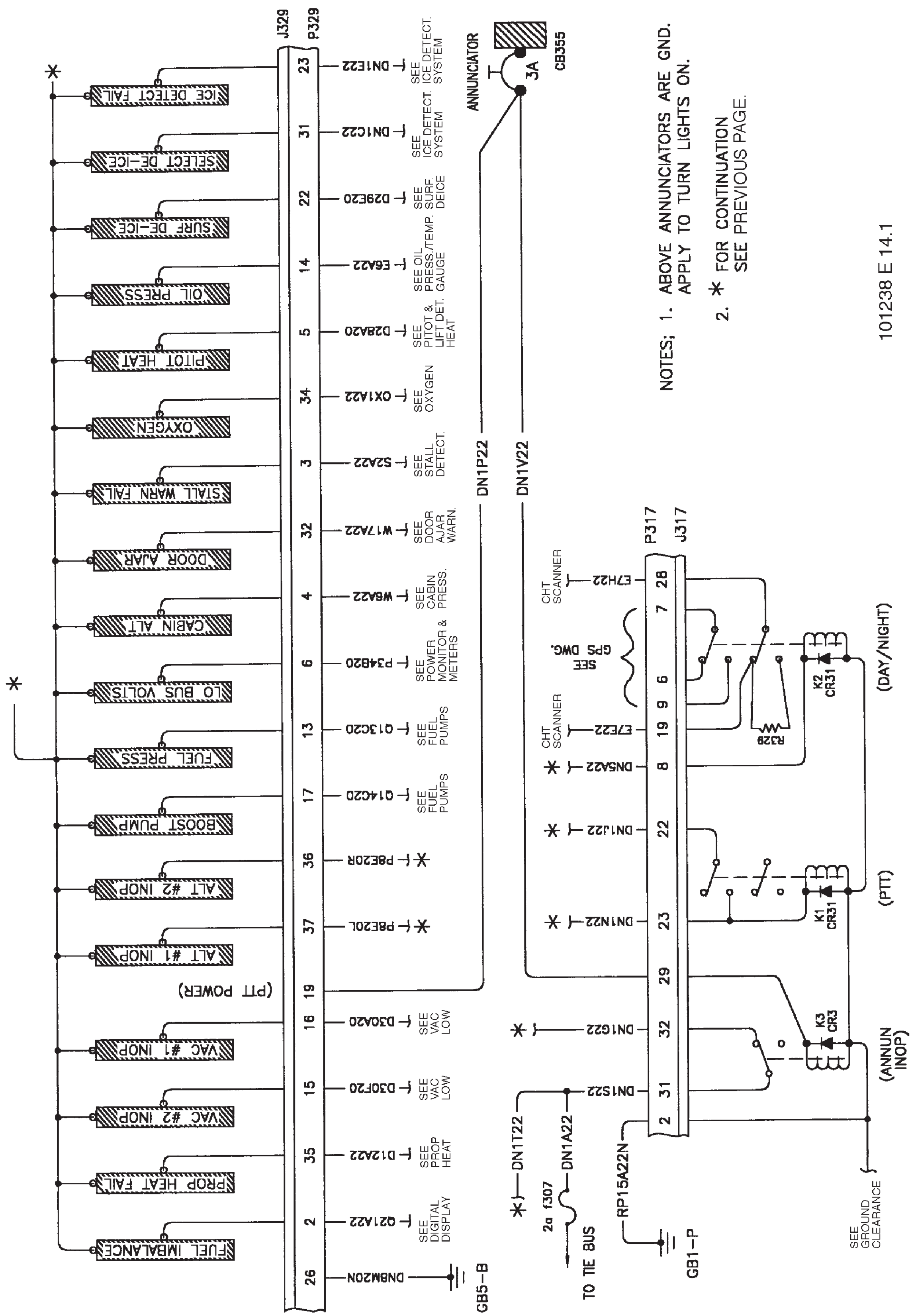
Annunciator Panel
Figure 1 (Sheet 4 of 18)



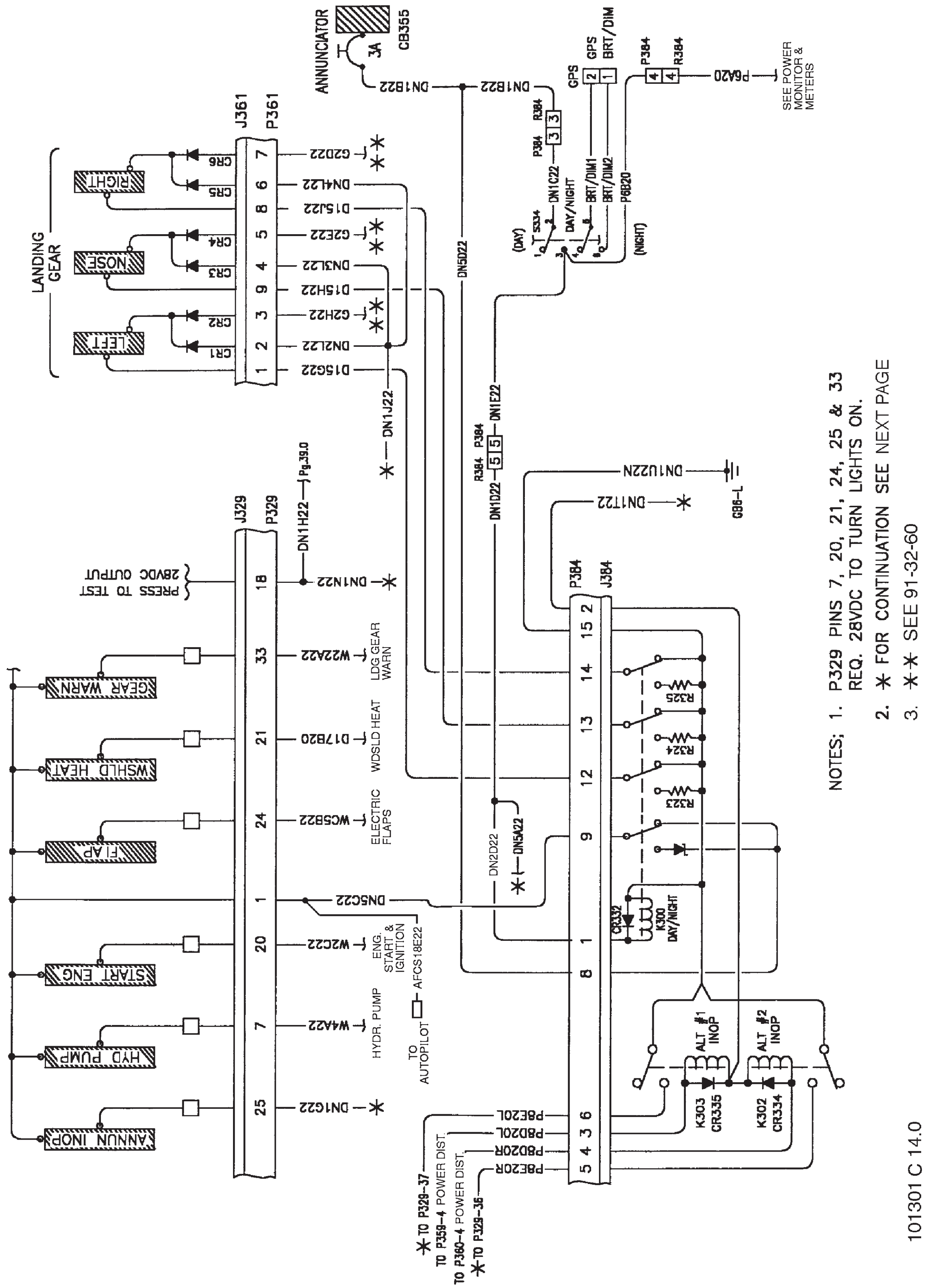
Annunciator Panel
Figure 1 (Sheet 5 of 18)



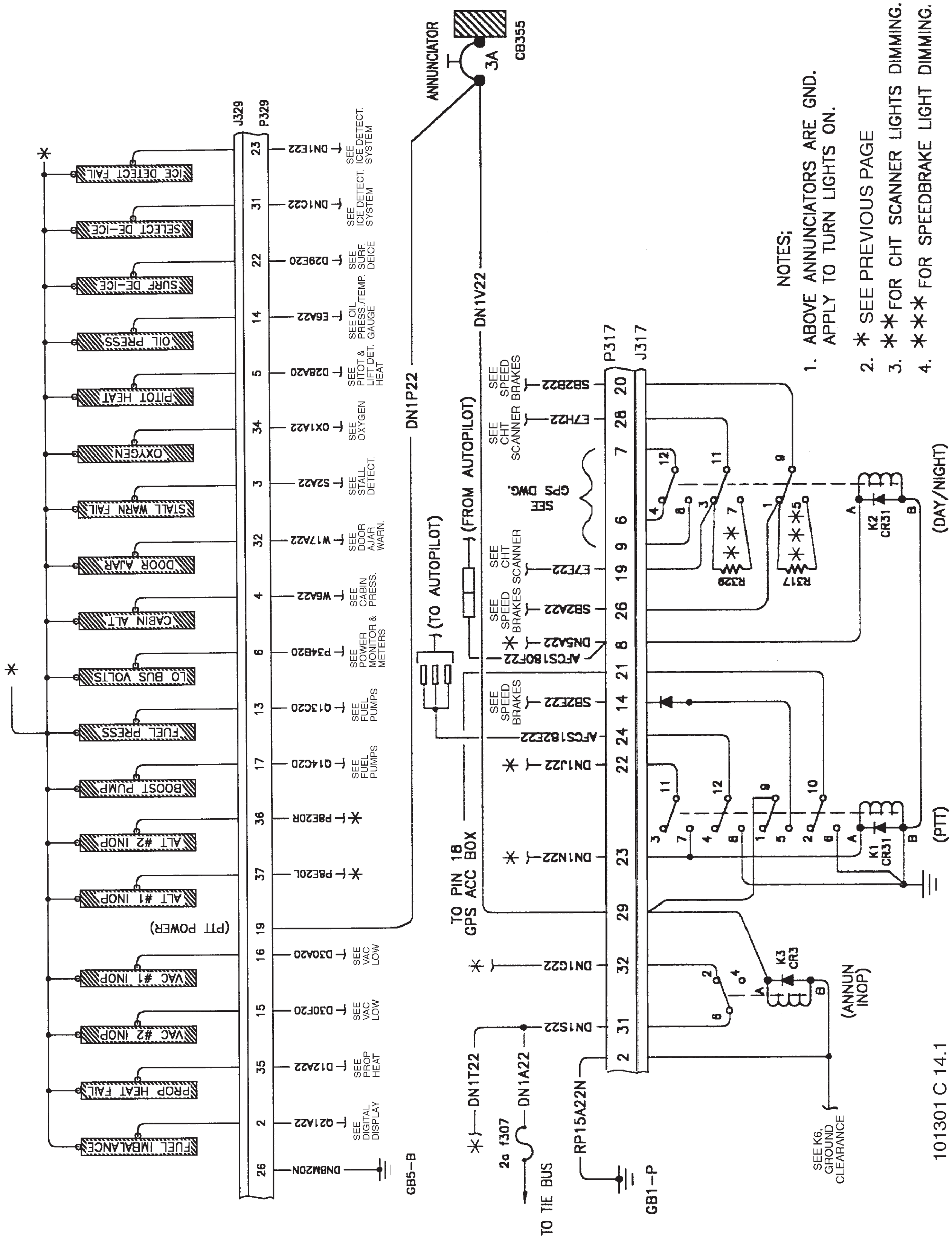
Annunciator Panel
Figure 1 (Sheet 6 of 18)



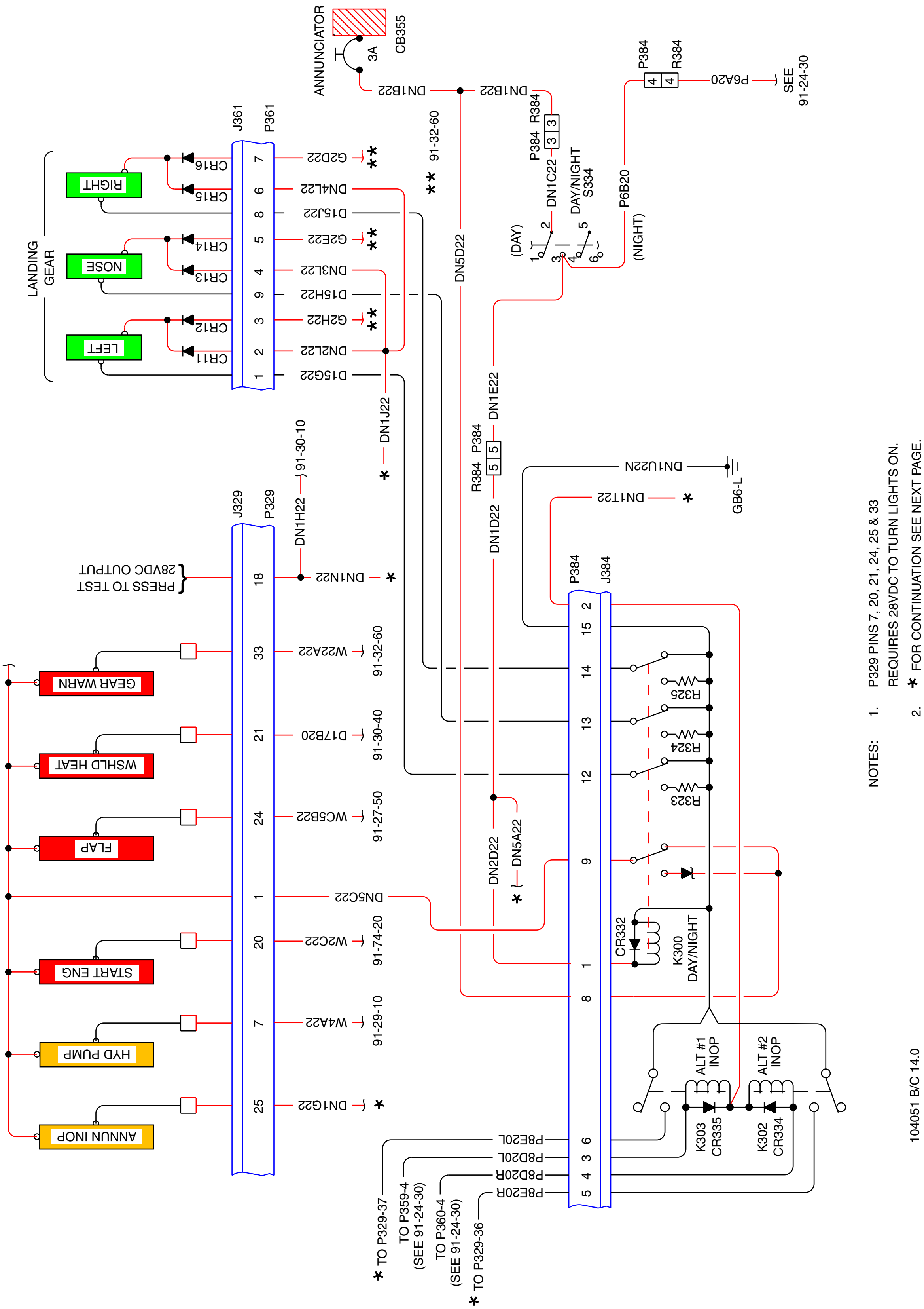
Annunciator Panel
Figure 1 (Sheet 7 of 18)



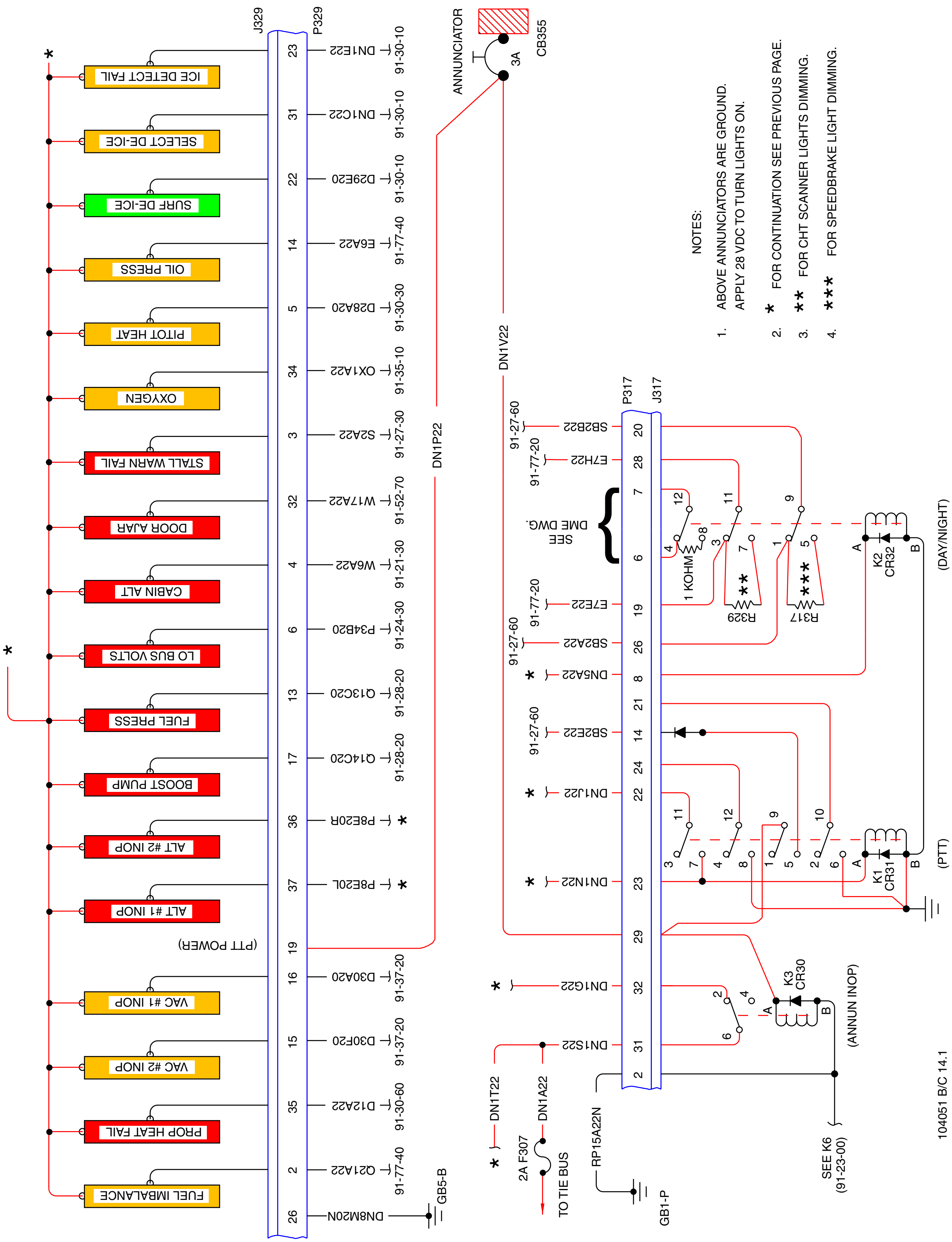
Annunciator Panel
Figure 1 (Sheet 8 of 18)



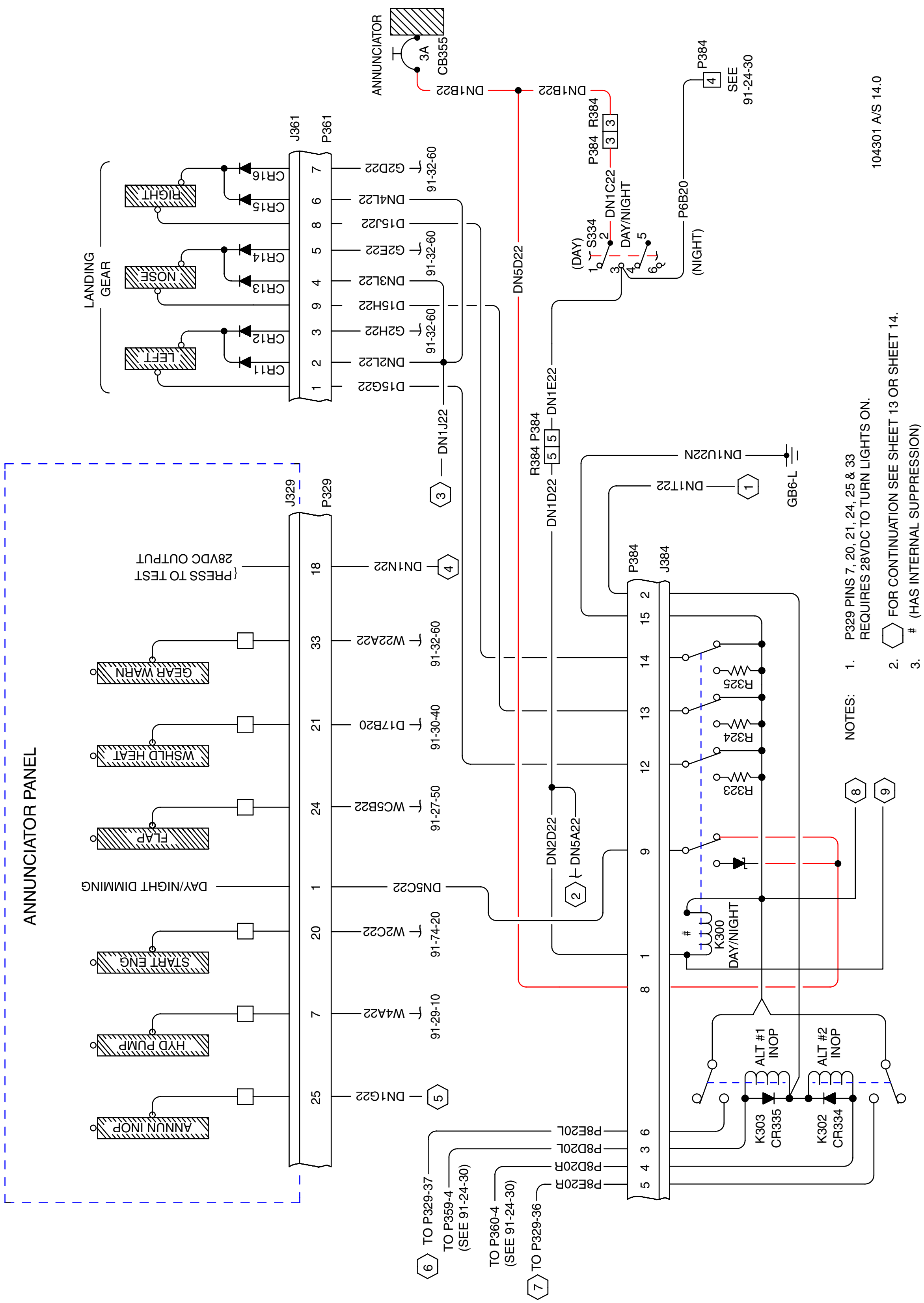
Annunciator Panel
Figure 1 (Sheet 9 of 18)



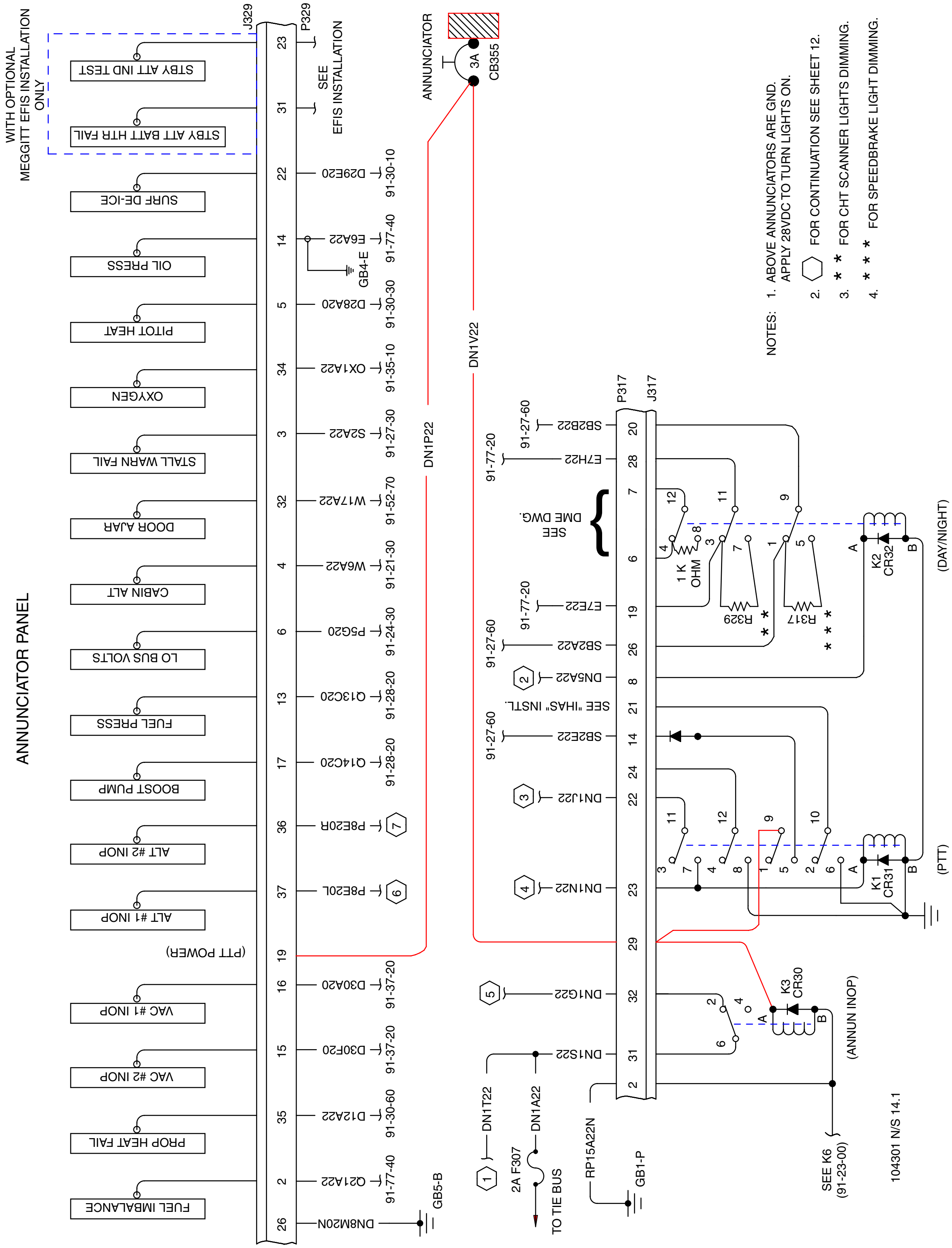
Annunciator Panel
Figure 1 (Sheet 10 of 18)



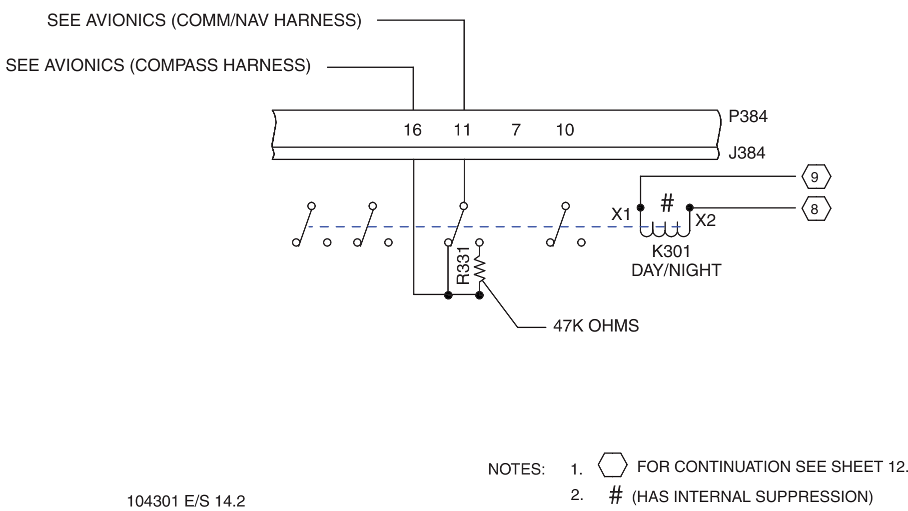
Annunciator Panel
Figure 1 (Sheet 11 of 18)



Annunciator Panel
Figure 1 (Sheet 12 of 18)



Annunciator Panel
Figure 1 (Sheet 13 of 18)



Annunciator Panel
Figure 1 (Sheet 14 of 18)

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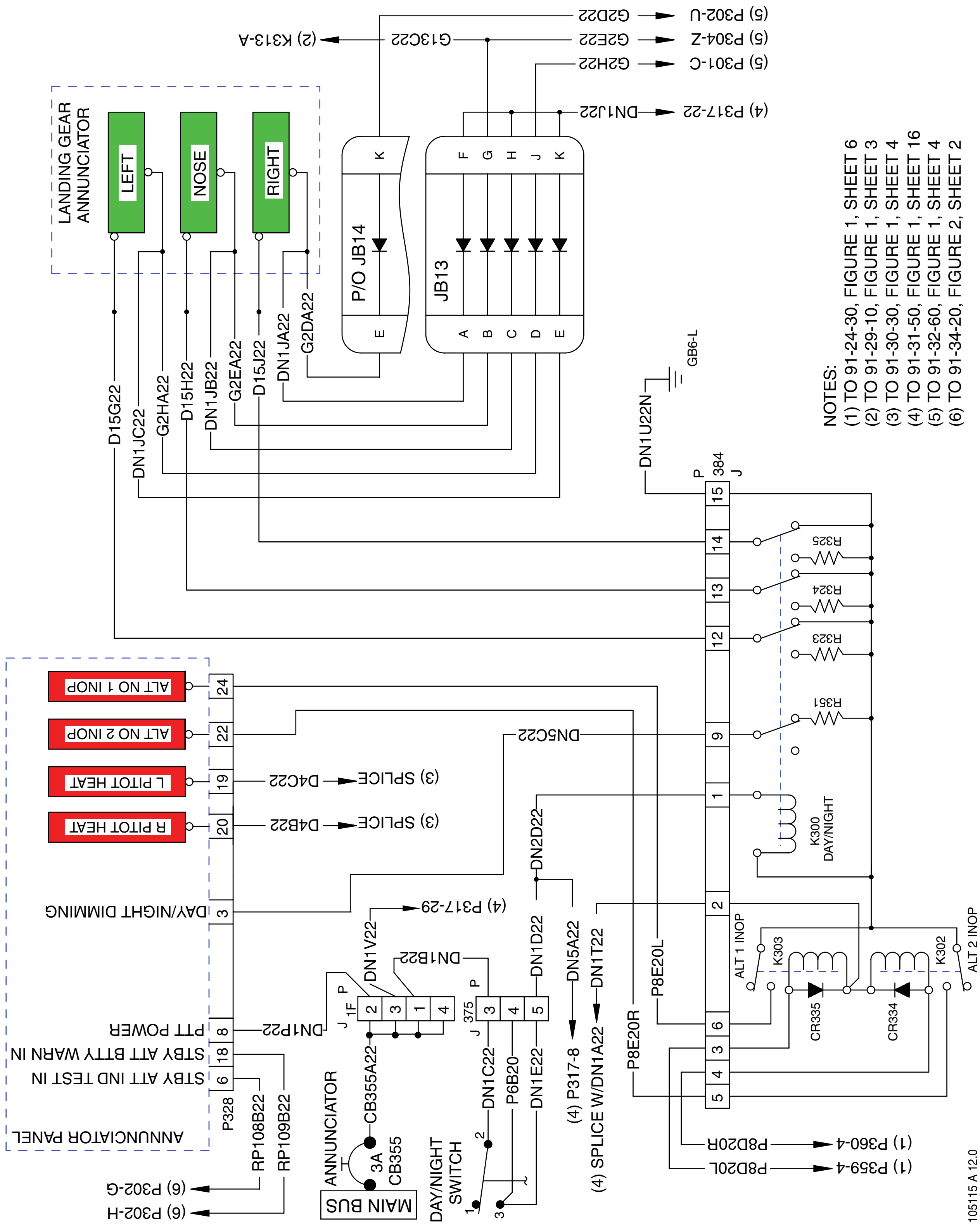
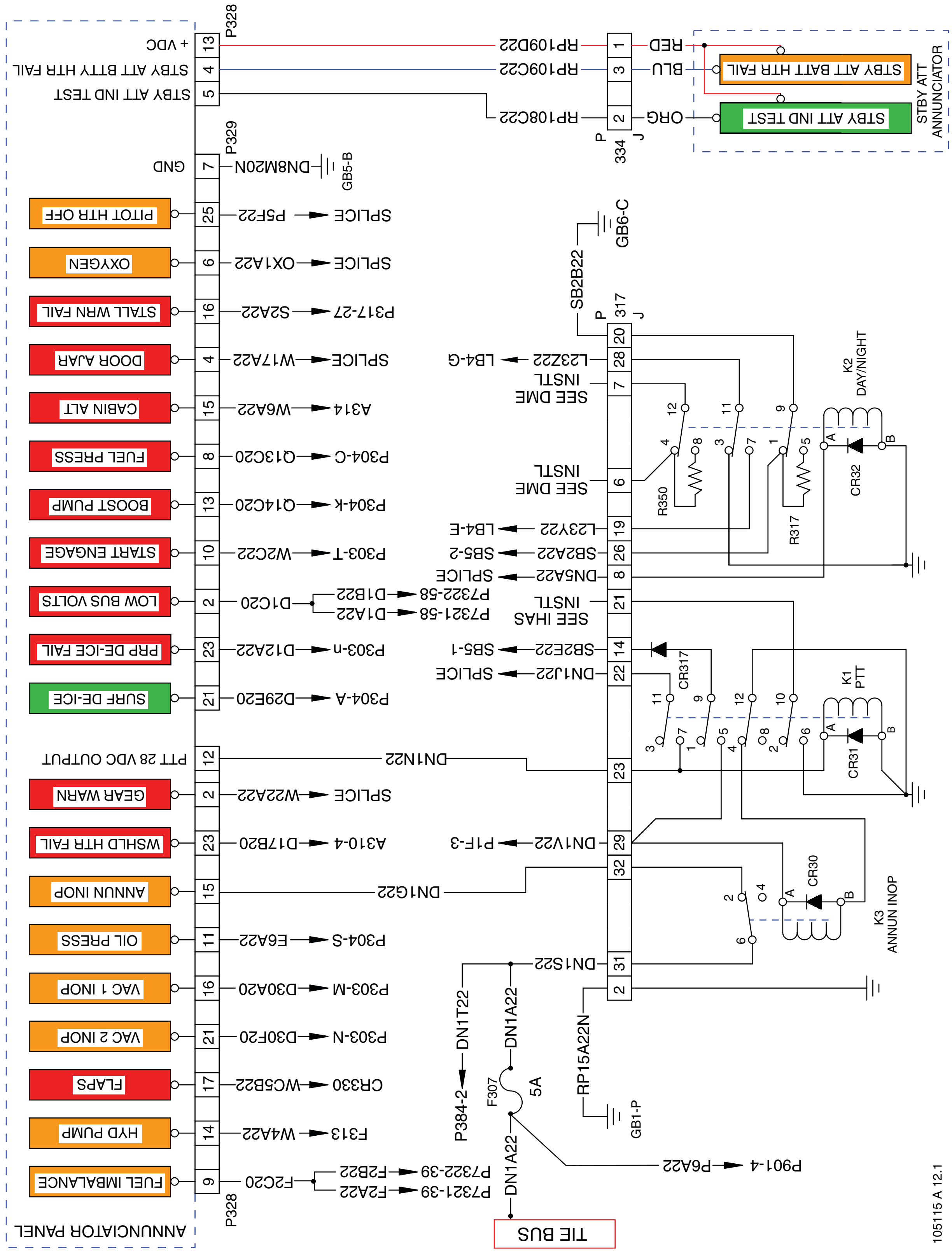
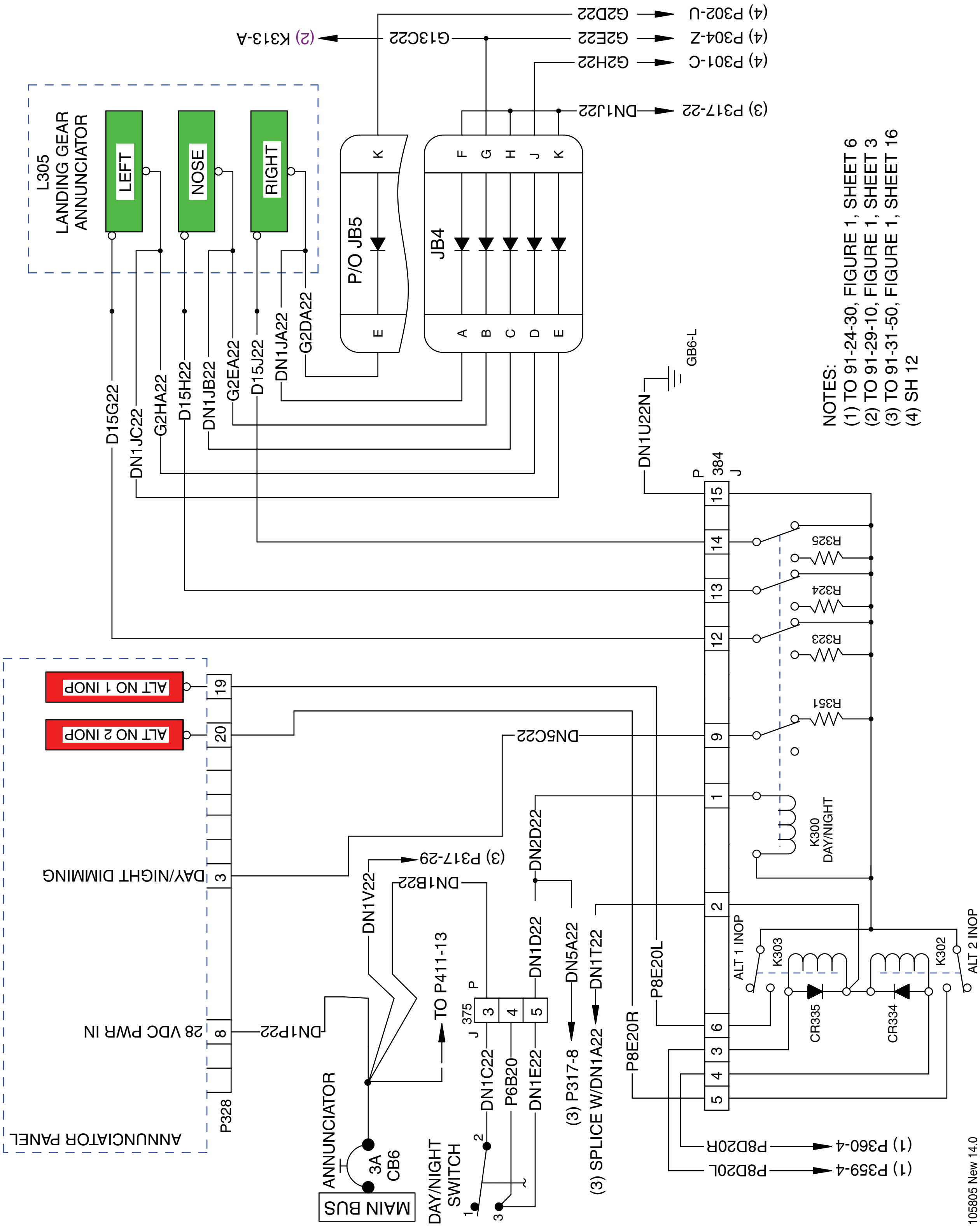


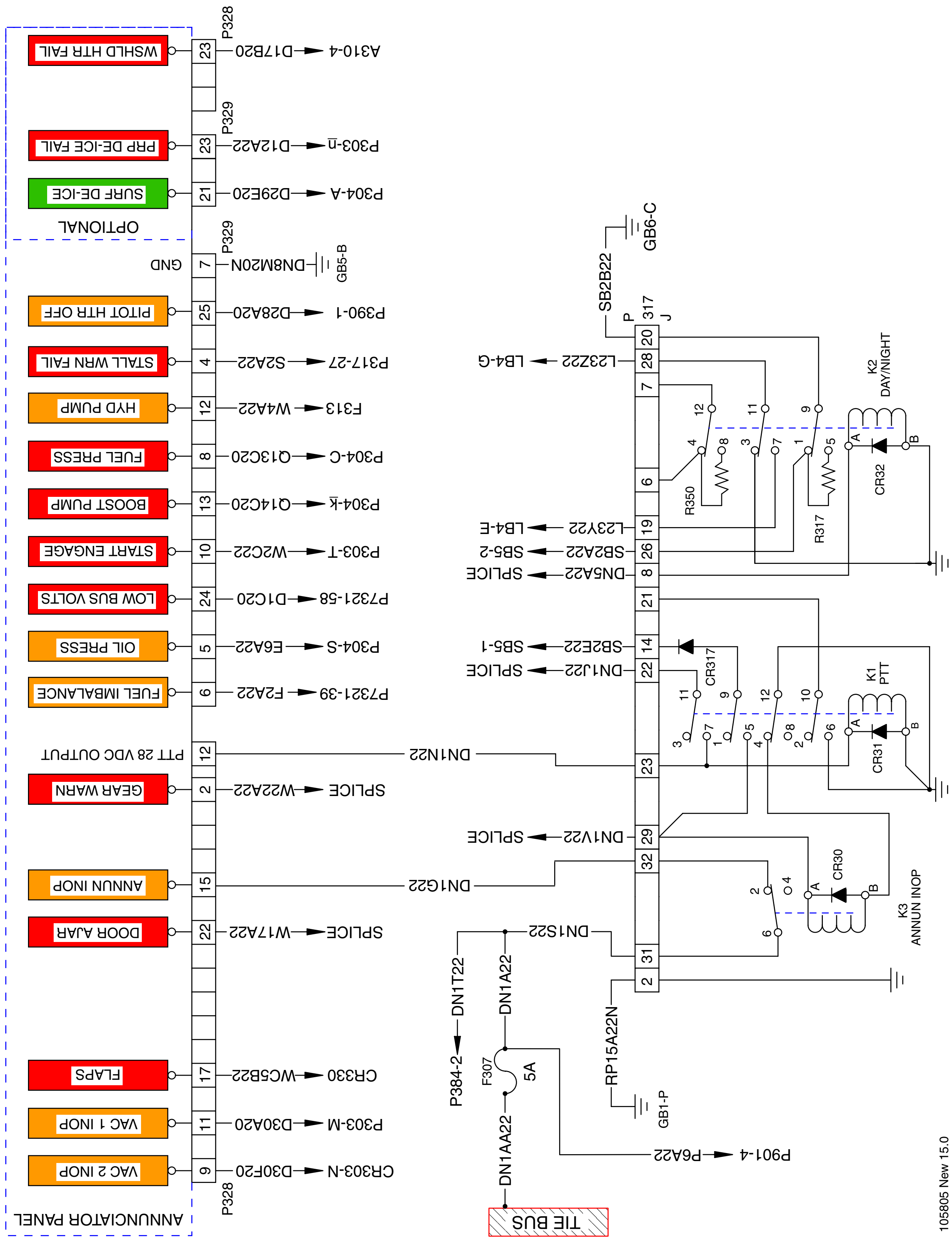
Annunciator Panel
Figure 1 (Sheet 15 of 18)



Annunciator Panel
Figure 1 (Sheet 16 of 18)



Annunciator Panel
Figure 1 (Sheet 17 of 18)



Annunciator Panel
Figure 1 (Sheet 18 of 18)

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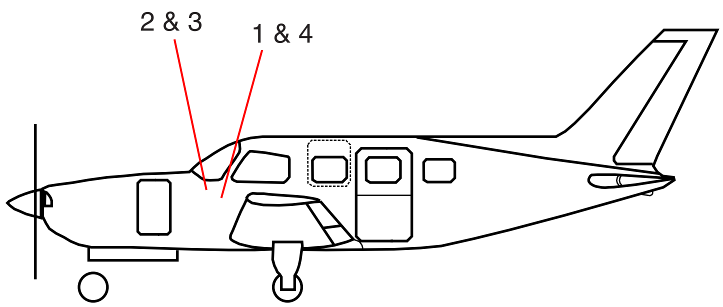
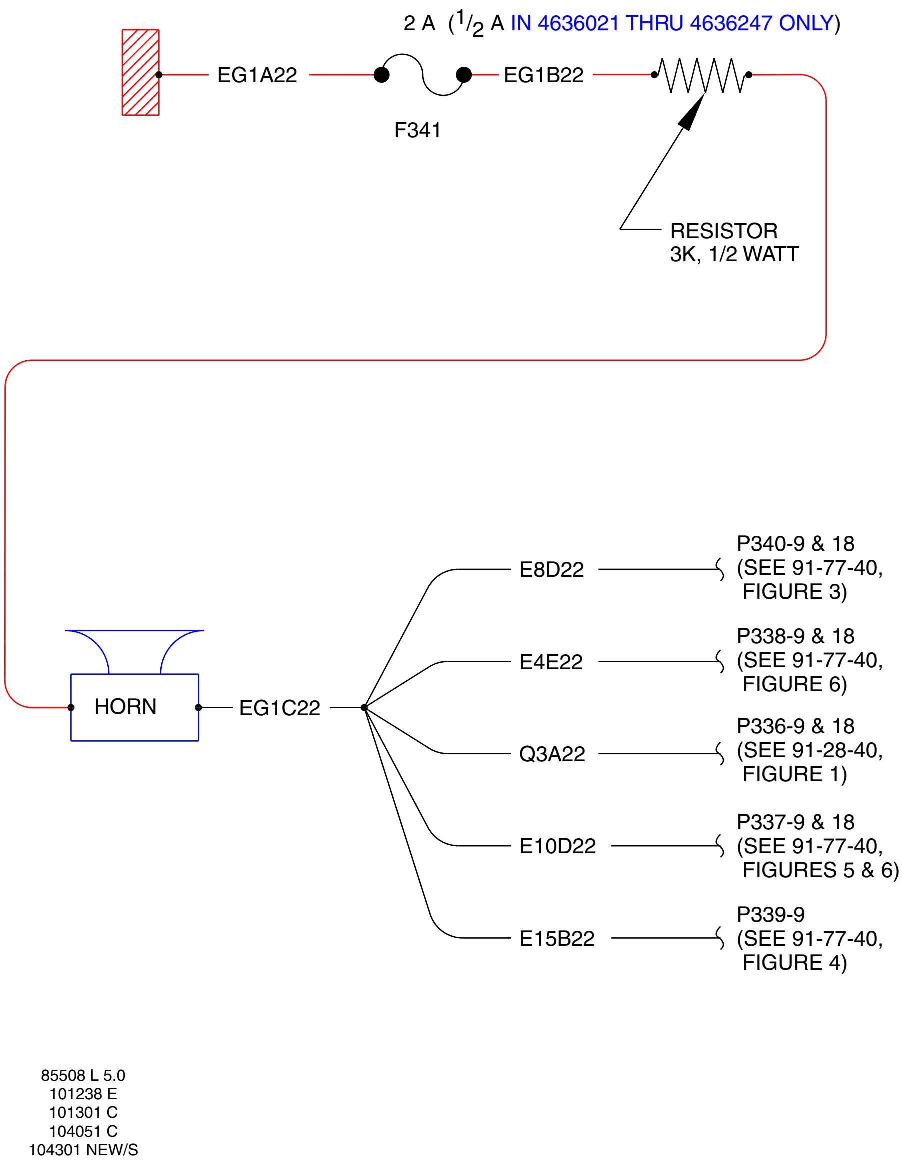
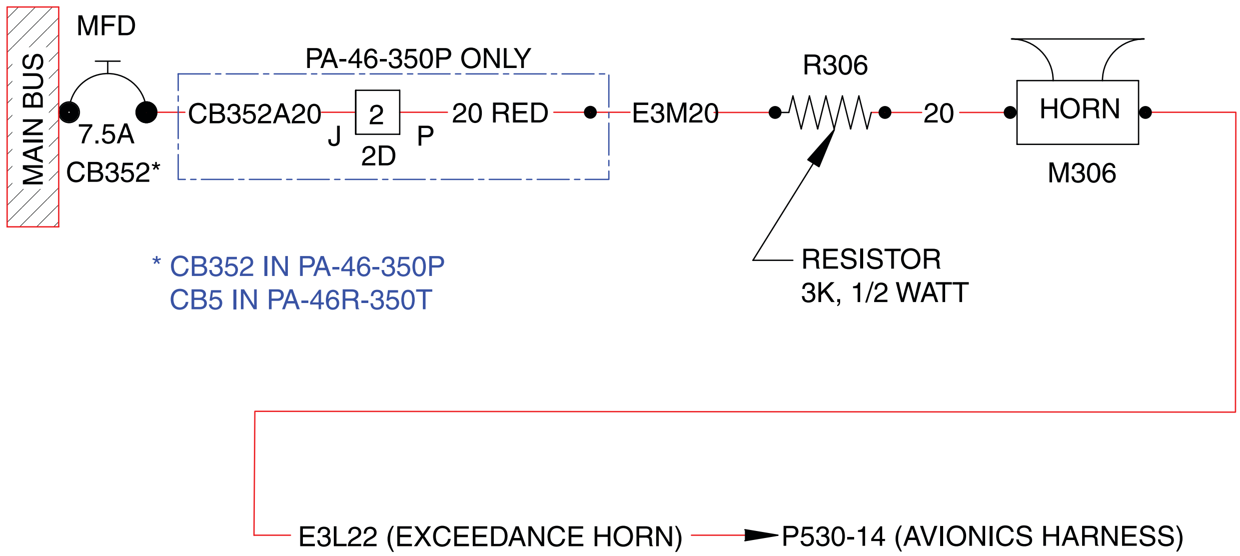
Item #	Designation	Description
1	F341	Fuse (2 Amp) (5 Amp with Avidyne Entegra) (½ Amp in S/N's 4636021 thru 4636247 only)
2		Resistor
3		Horn
4	CB352, CB5	Circuit Breaker - MFD (7.5 Amp with Avidyne Entegra)



Exceedance Audio Alert
Figure 2 (Sheet 1 of 3)



Exceedance Audio Alert
Figure 2 (Sheet 2 of 3)



Exceedance Audio Alert
Figure 2 (Sheet 3 of 3)

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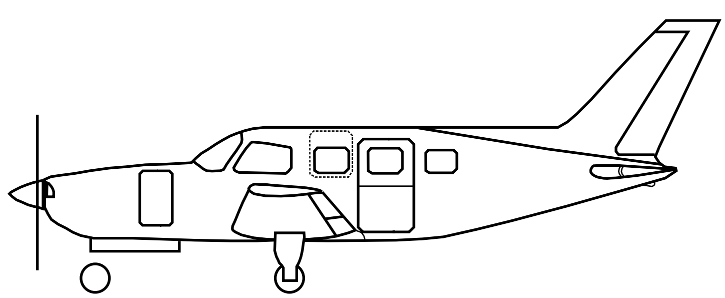
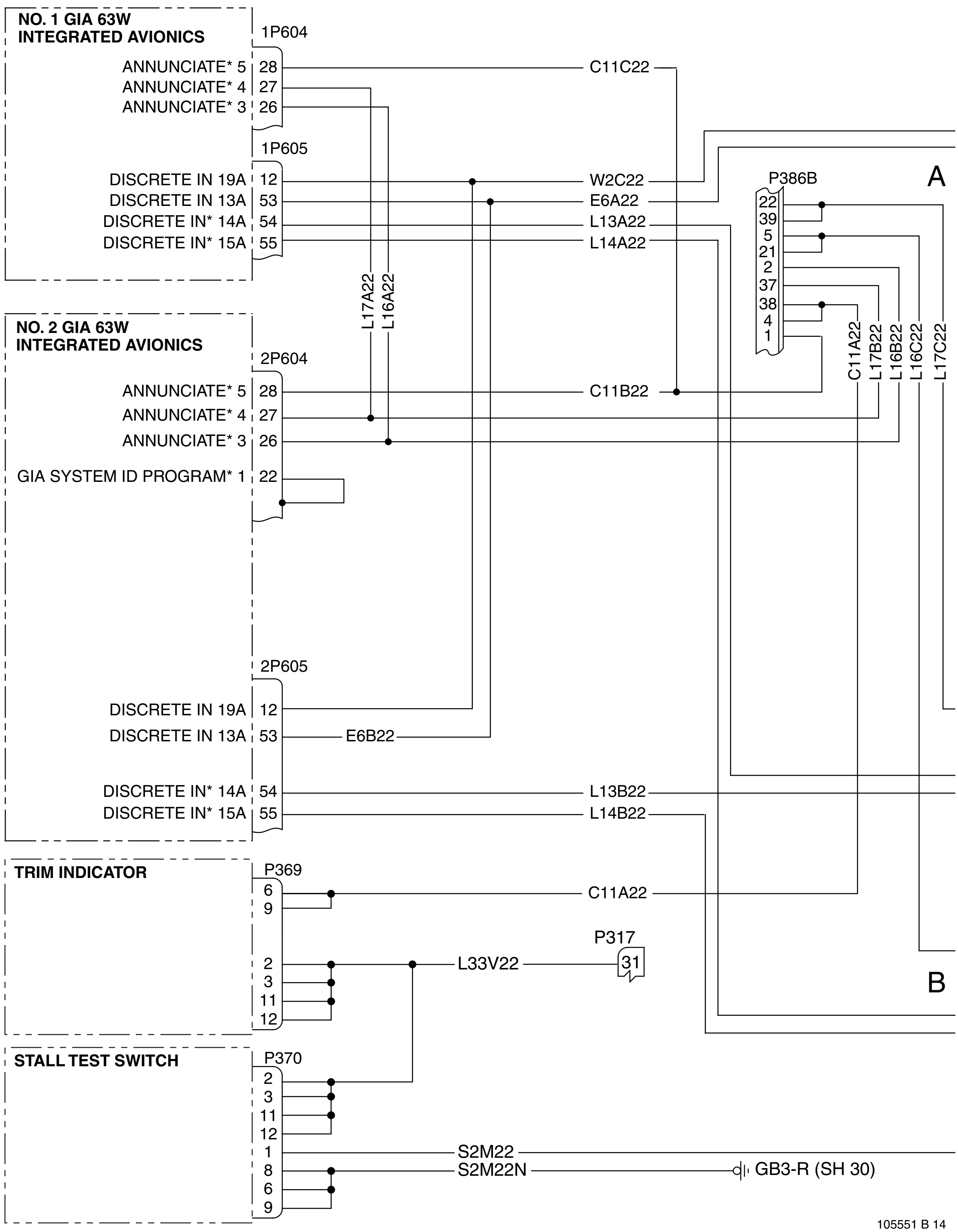
Item #	Designation	Description

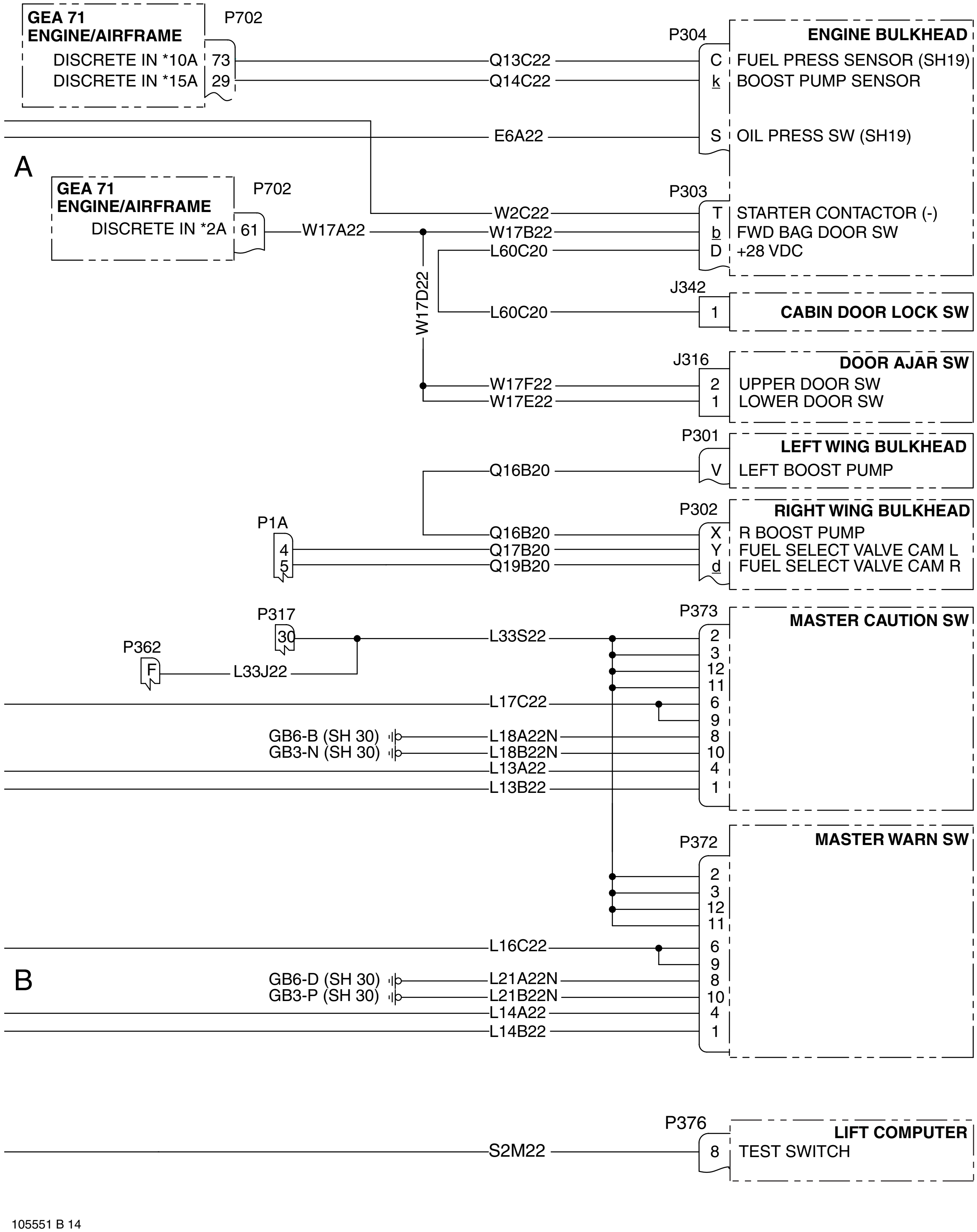
Information Pending - See Parts Catalog



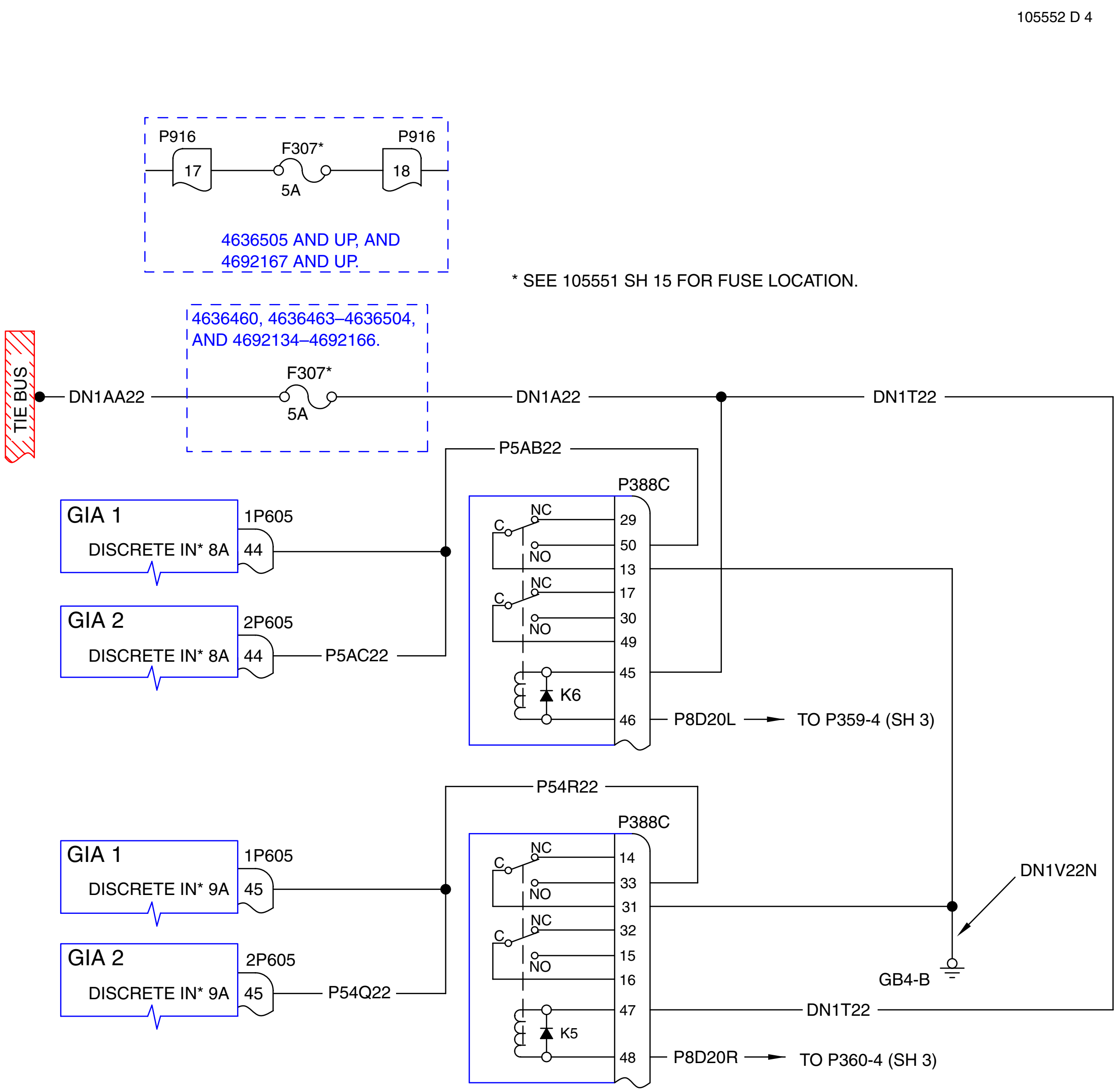
Crew Alerting System
Figure 3 (Sheet 1 of 3)



Crew Alerting System
Figure 3 (Sheet 2 of 3)

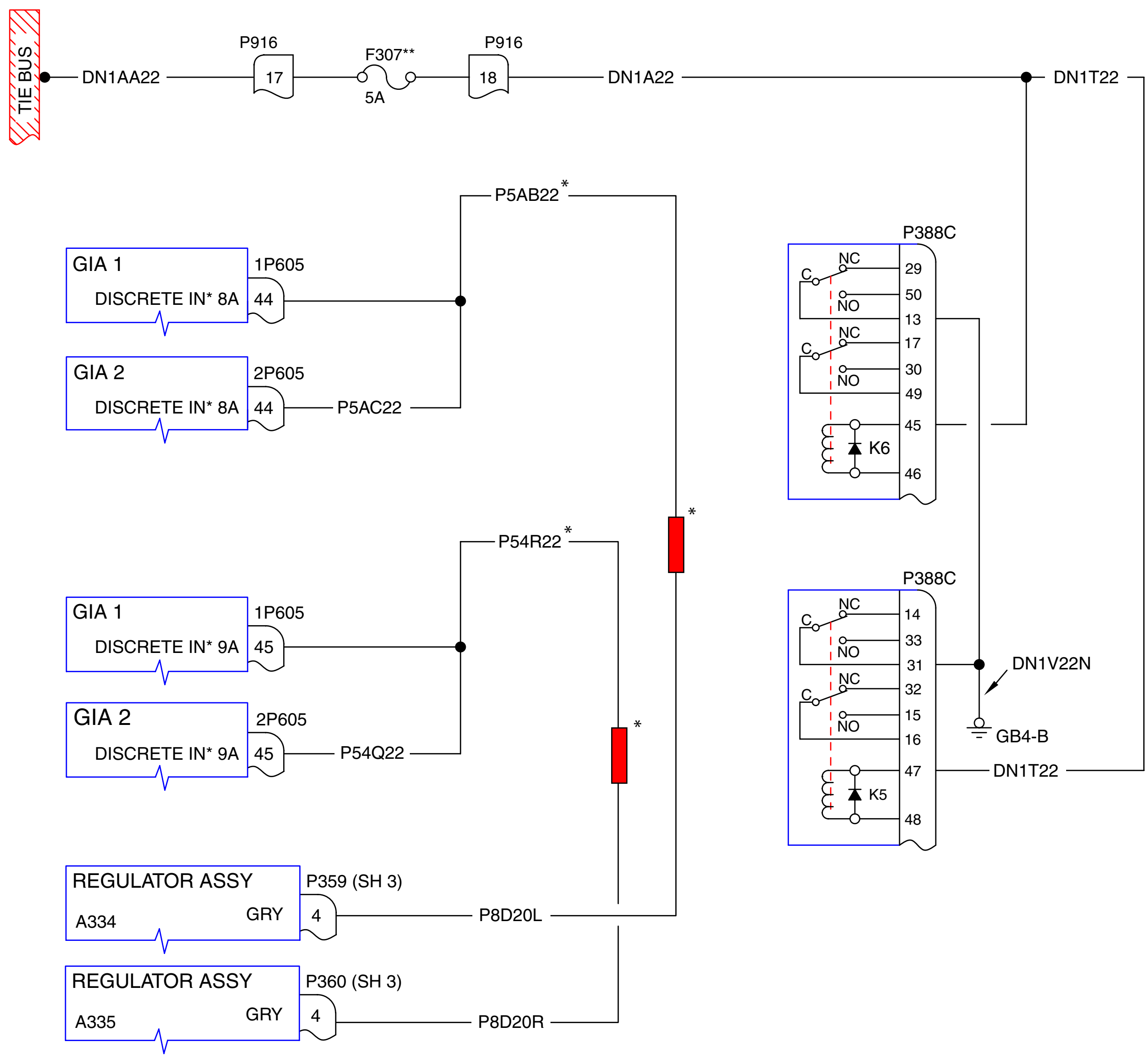


Crew Alerting System
Figure 3 (Sheet 3 of 3)



Alternator System Fail Annunciation
Figure 4 (Sheet 1 of 2)

105552 4



* Wire codes and splices only in airplanes modified per Piper SL 1168.
** See 105551 SH 15 for fuse location.

Alternator System Fail Annunciation
Figure 4 (Sheet 2 of 2)

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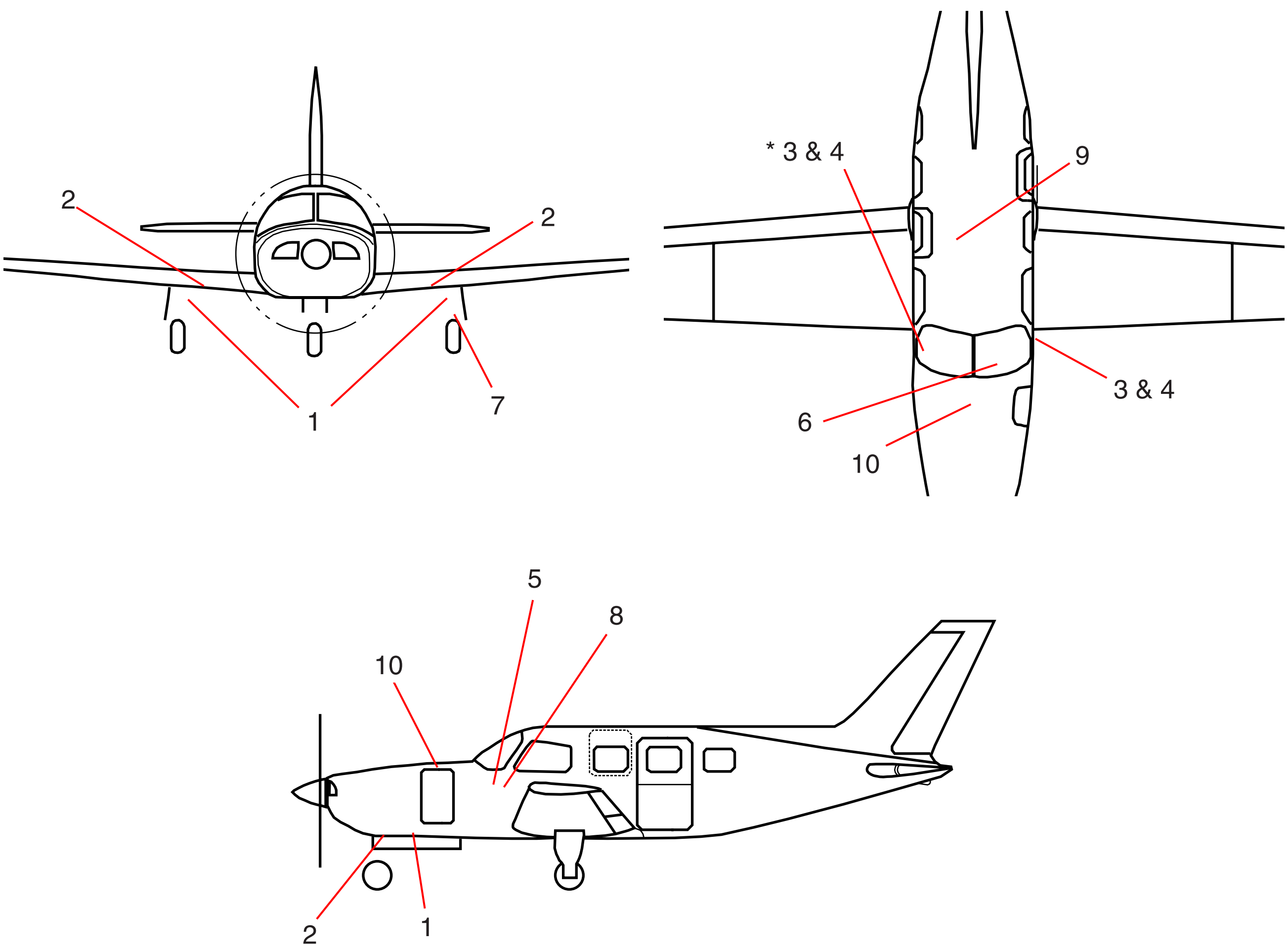
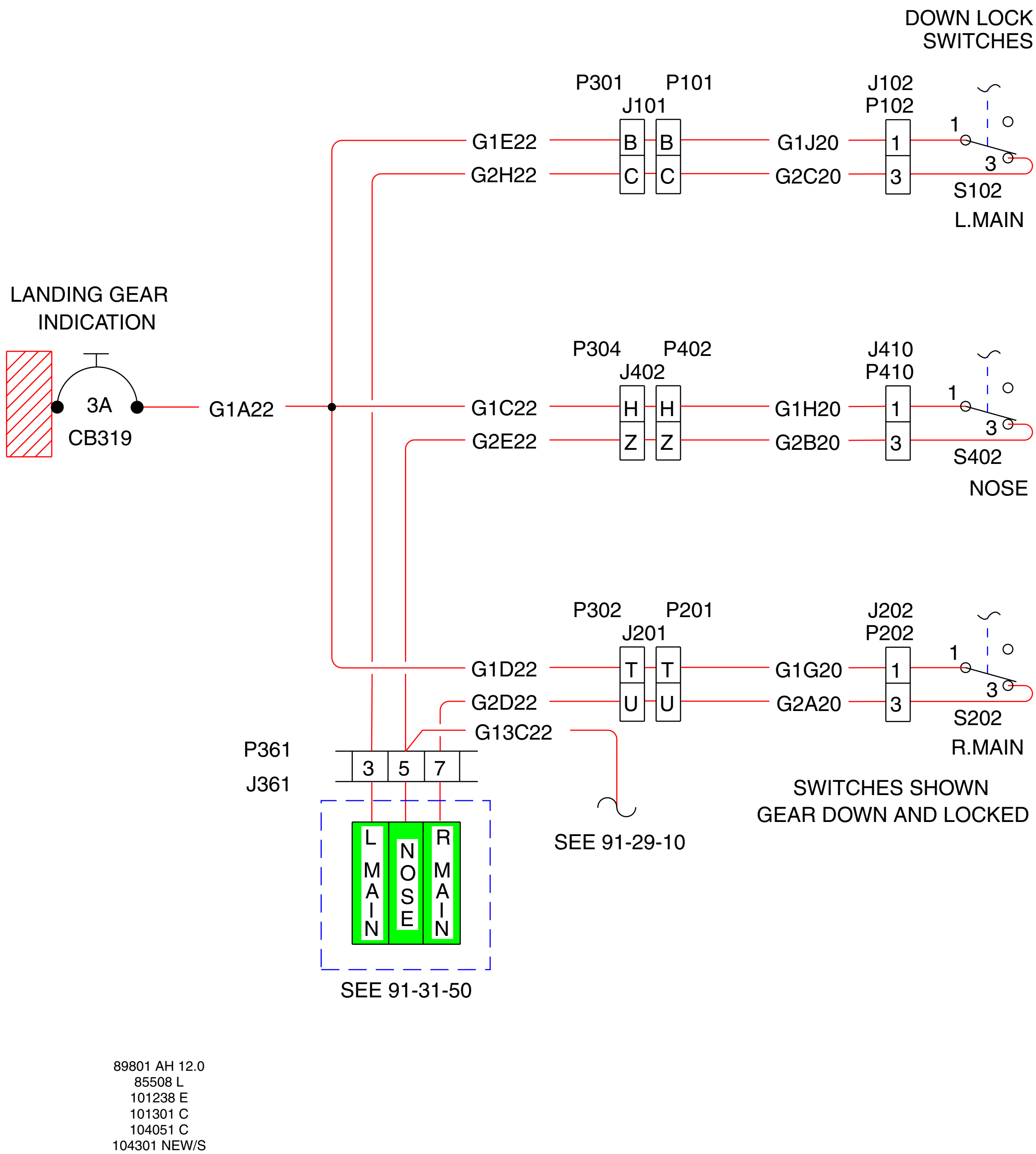
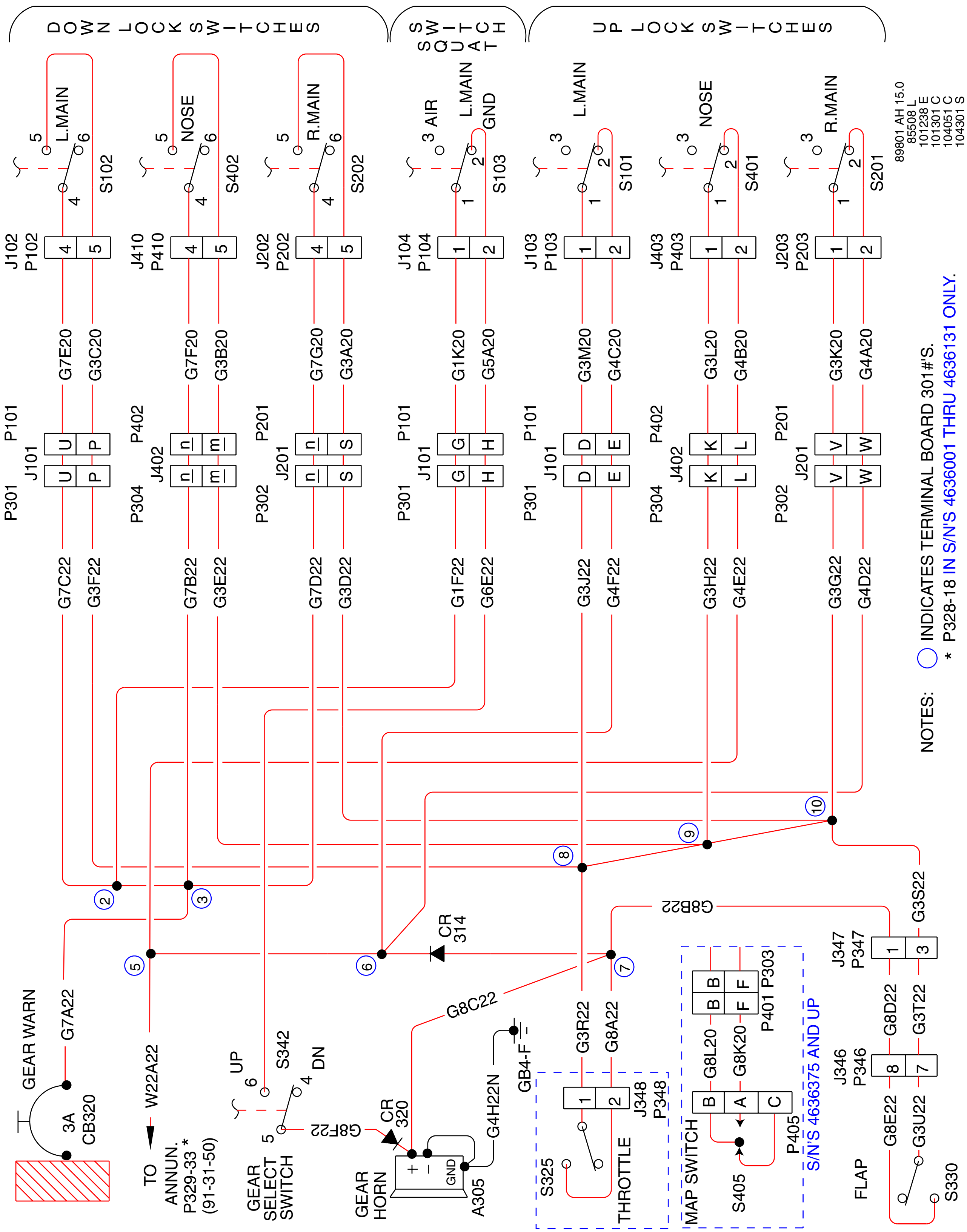
Item #	Designation	Description
1	S102, S202, S402	Left, Right Main and Nose Gear Down Lock Switch Assembly
2	S101, S201, S401	Left, Right Main and Nose Gear Up Lock Switch Assembly
3	CB319, CB24*	Circuit Breaker - Landing Gear Indicator (3 Amp)
4	CB320, CB25*	Circuit Breaker - Gear Warn (3 Amp)
5	S342	Gear Selector Switch
6	A305	Dual Warning Horn
7	S103	Squat Switch Assembly
8	S325	Throttle Switch Assembly (S/N's 4636001 thru 4636374 only)
9	S330	Flap Switch
10	S405	MAP Switch (S/N's 4636375 and up)



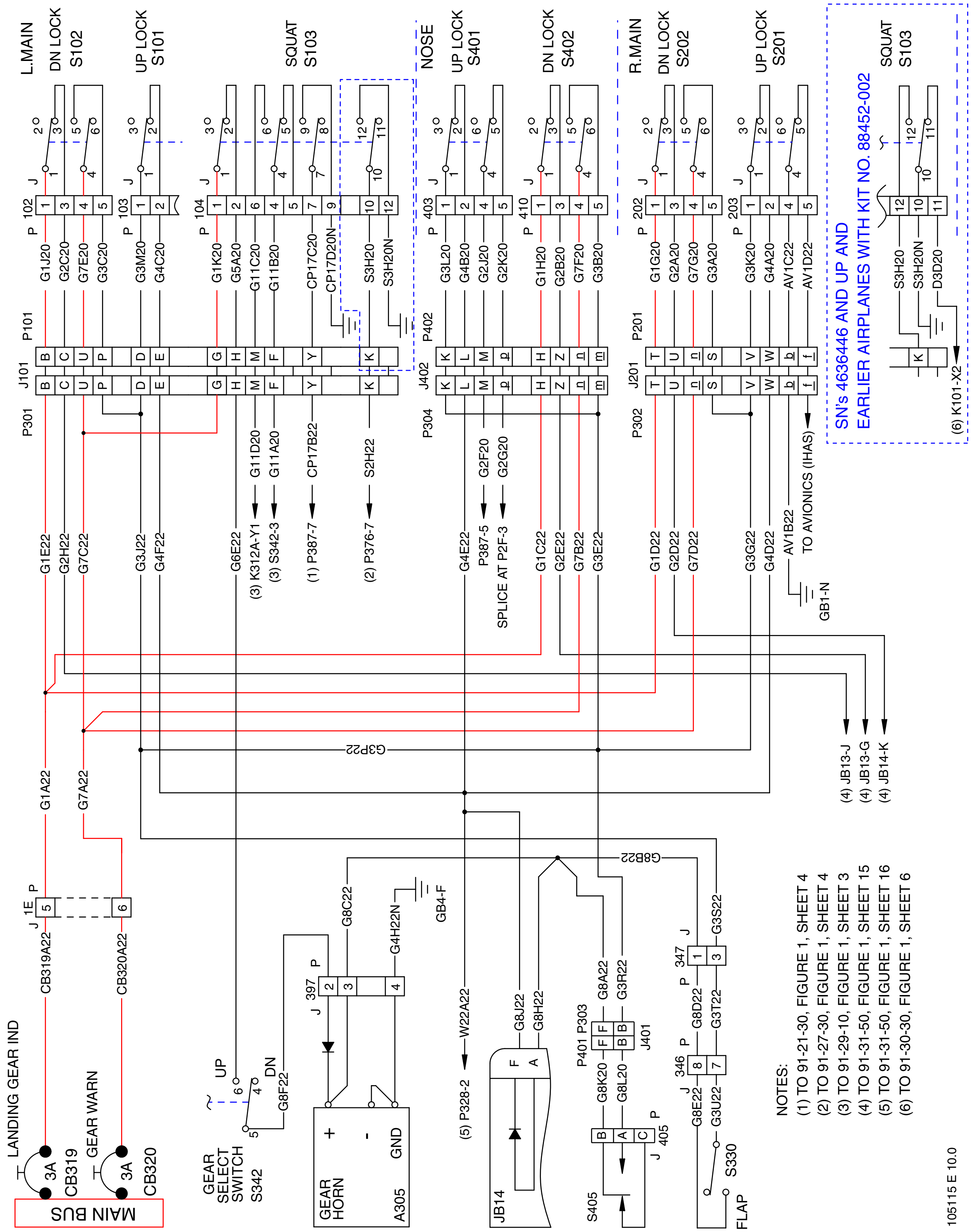
Landing Gear Position and Warning
Figure 1 (Sheet 1 of 9)



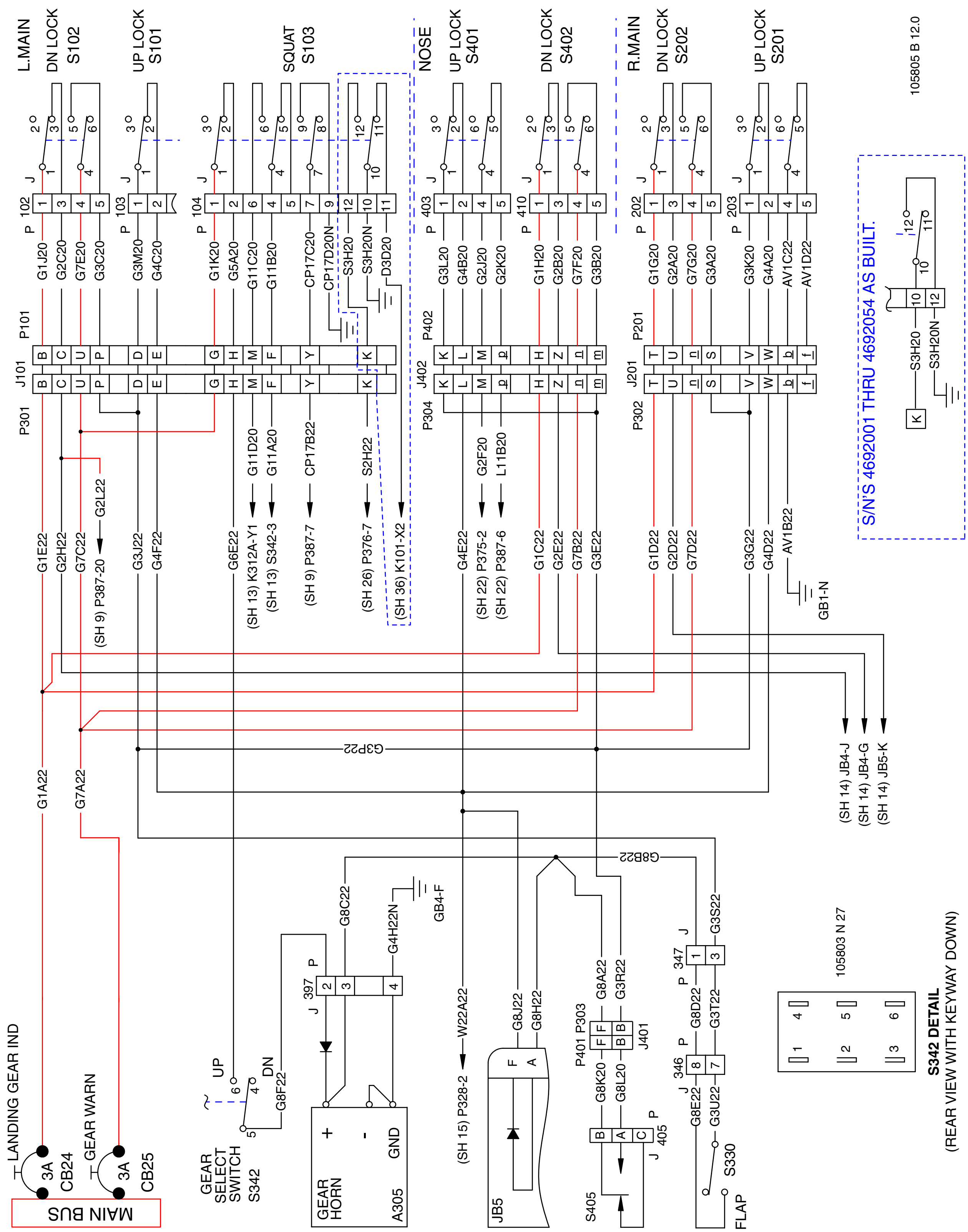
Landing Gear Position and Warning
Figure 1 (Sheet 2 of 9)



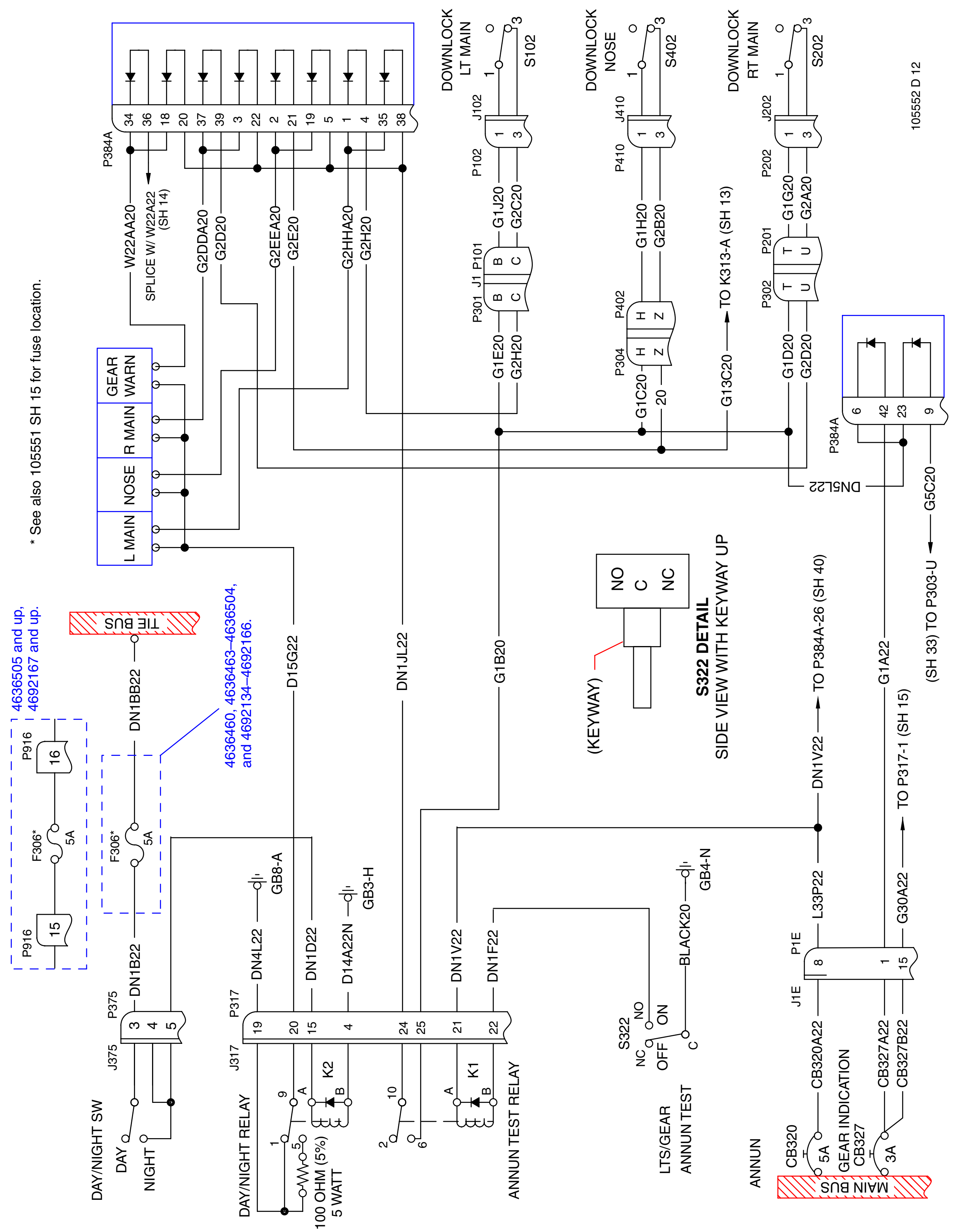
Landing Gear Position and Warning
Figure 1 (Sheet 3 of 9)



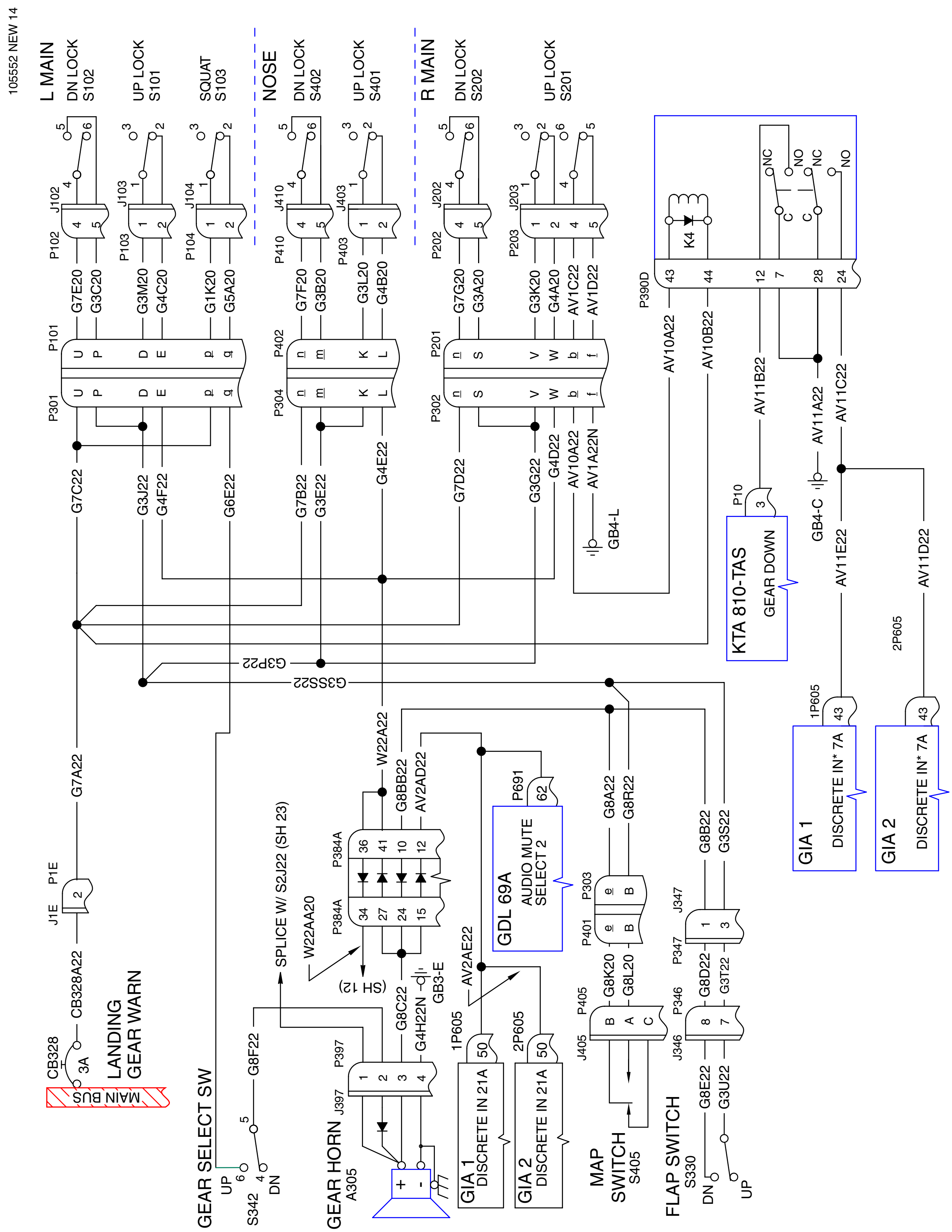
Landing Gear Position and Warning
Figure 1 (Sheet 4 of 9)



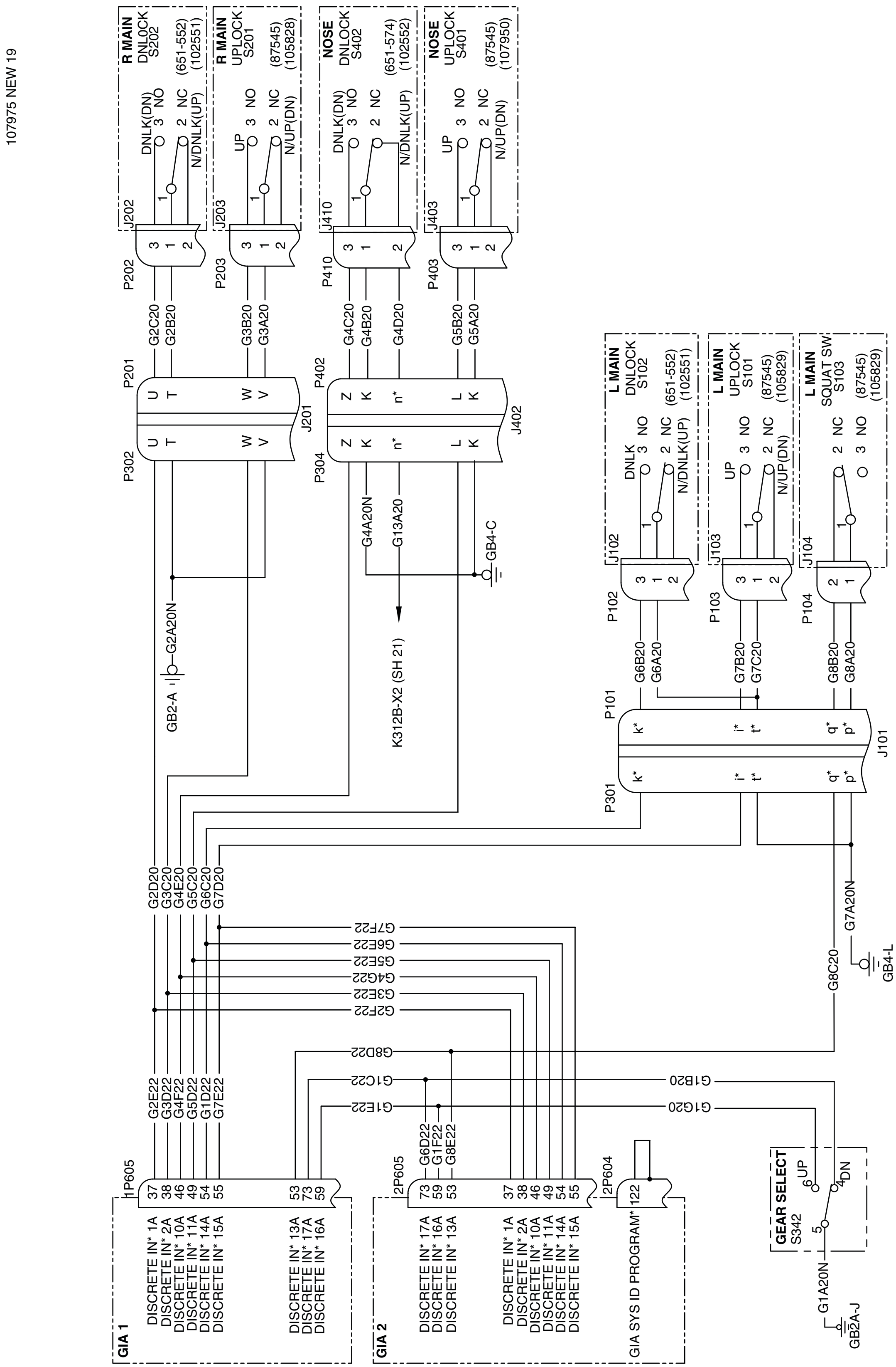
Landing Gear Position and Warning
Figure 1 (Sheet 5 of 9)



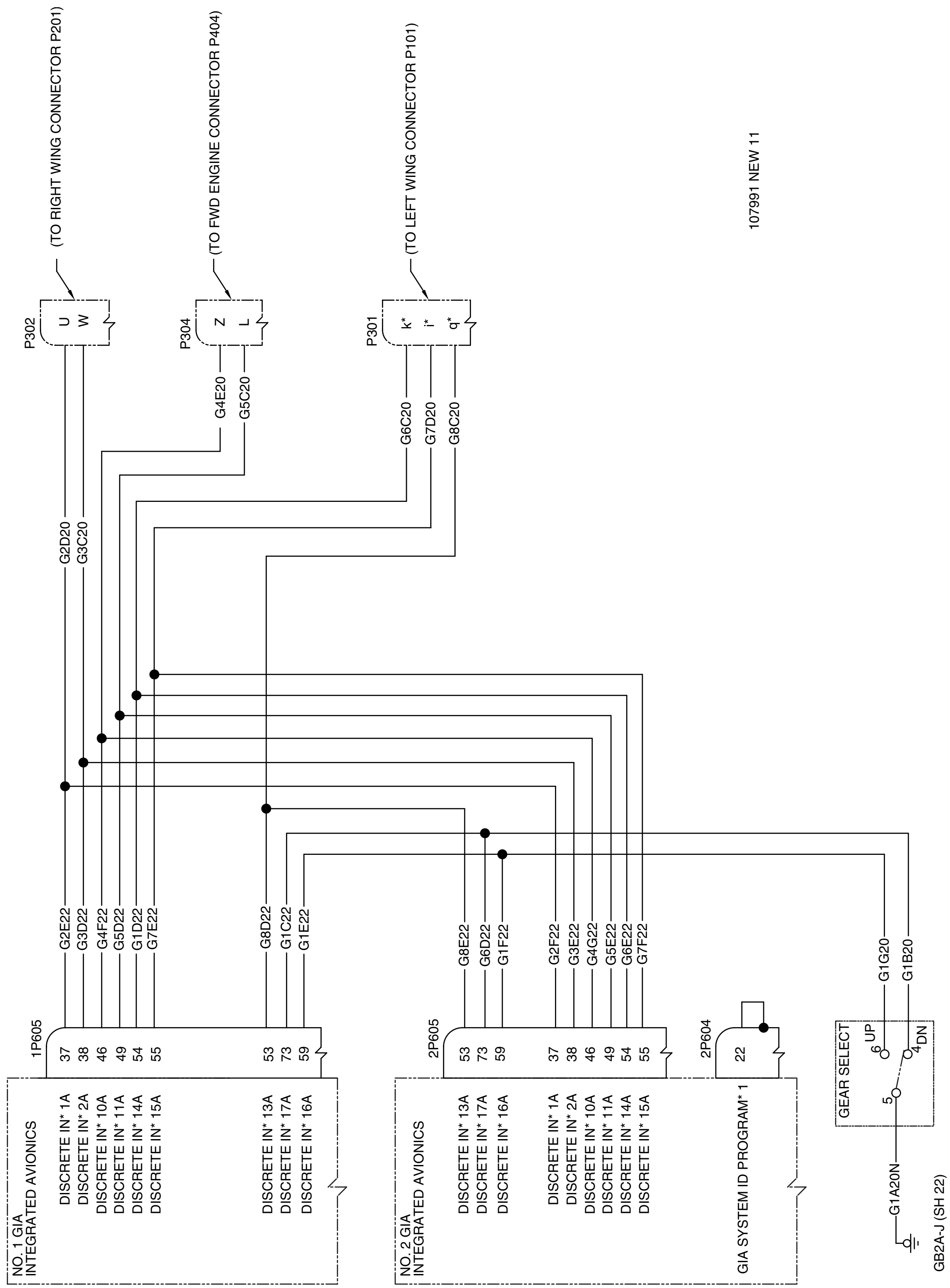
Landing Gear Position and Warning
Figure 1 (Sheet 6 of 9)



Landing Gear Position and Warning
Figure 1 (Sheet 7 of 9)



Landing Gear Position and Warning
Figure 1 (Sheet 8 of 9)



Landing Gear Position and Warning
Figure 1 (Sheet 9 of 9)

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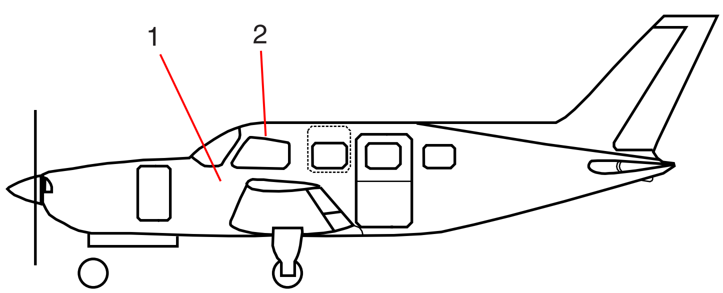
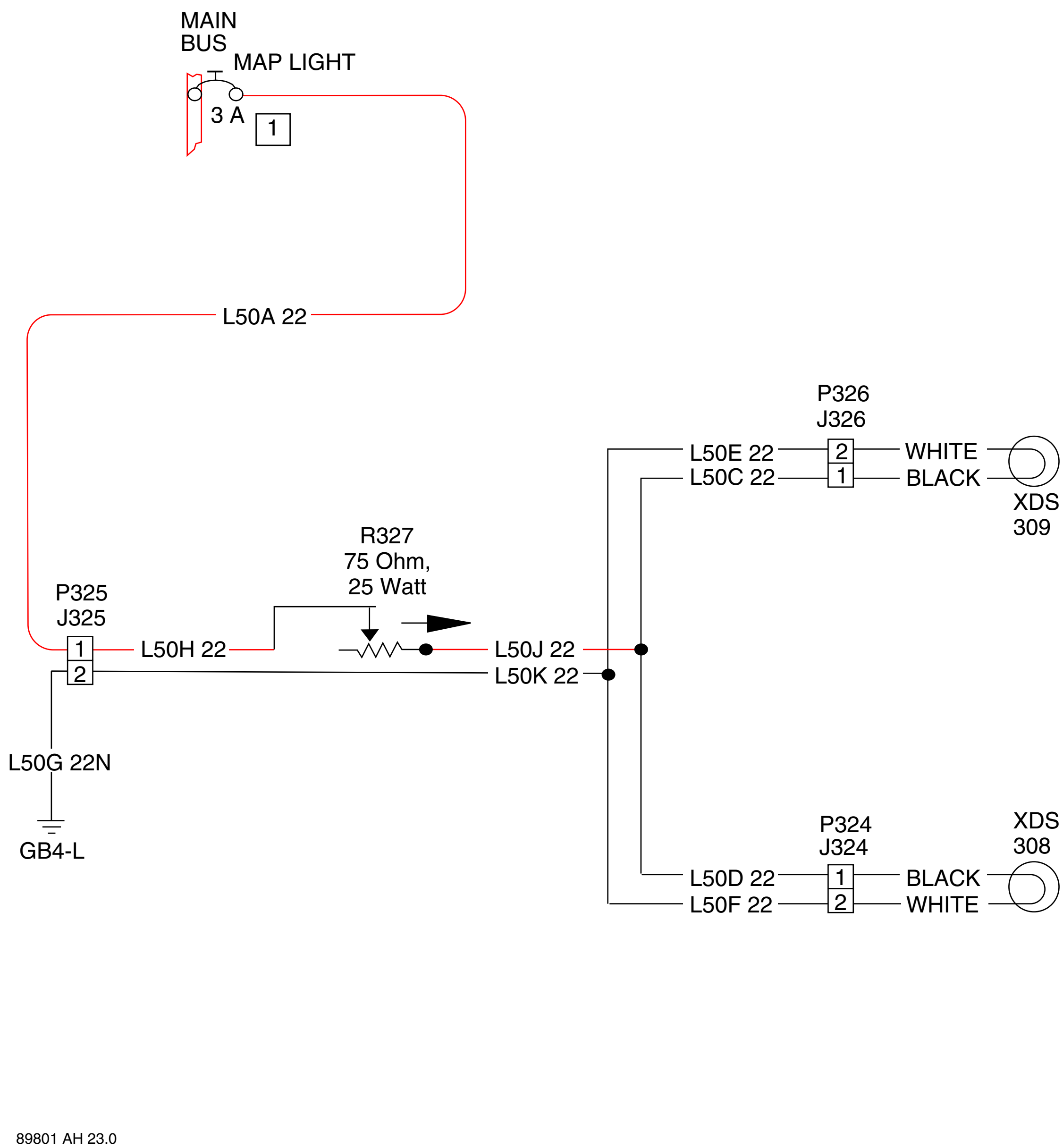
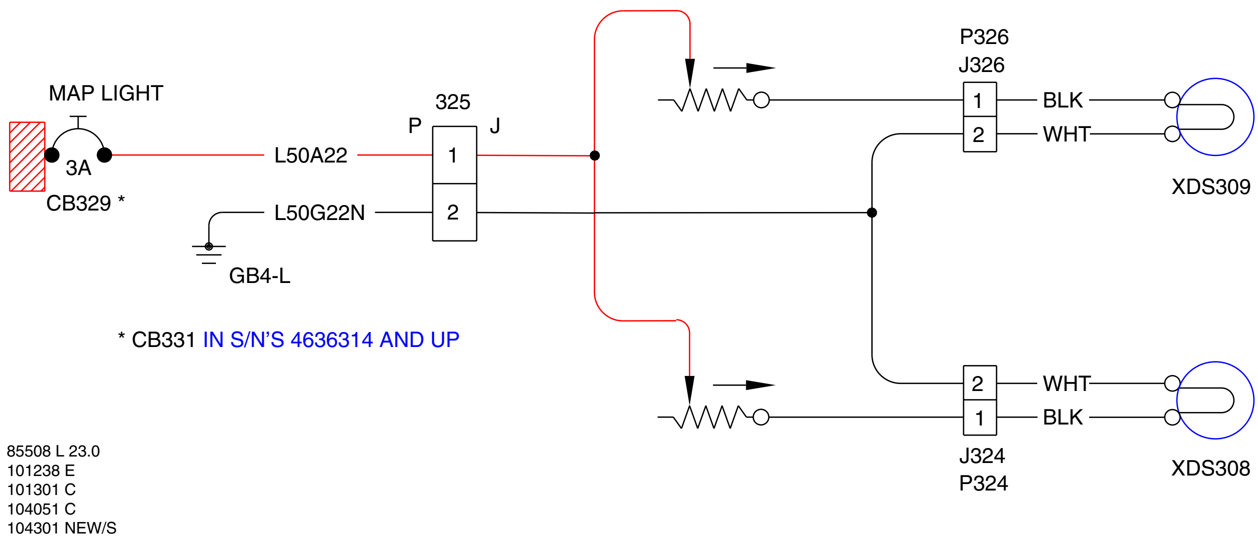
Item #	Designation	Description
1	CB329, CB331, CB56	Circuit Breaker - Map Light (3 Amp)
2	XDS308, XDS309	Map Light

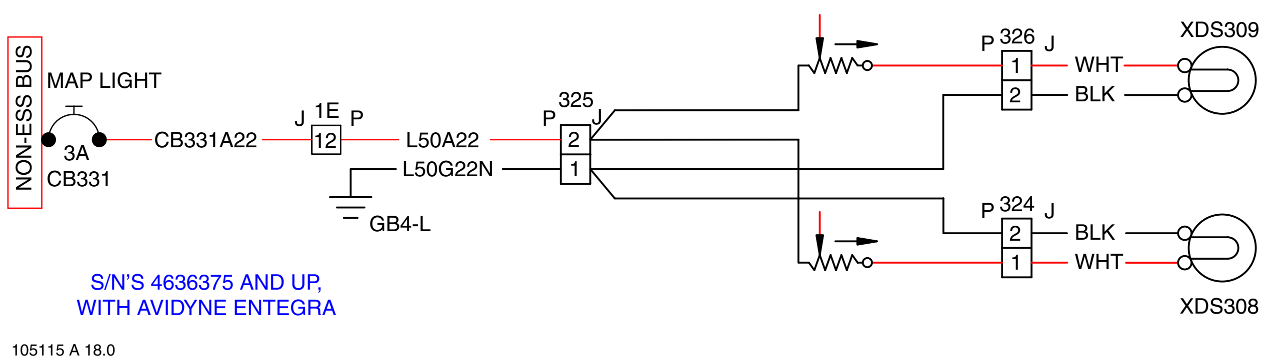


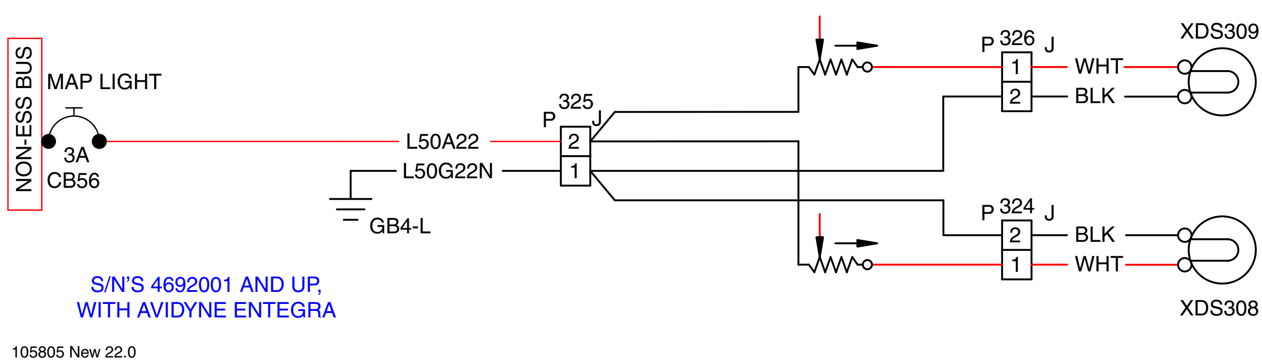
Map Lights
Figure 1 (Sheet 1 of 4)



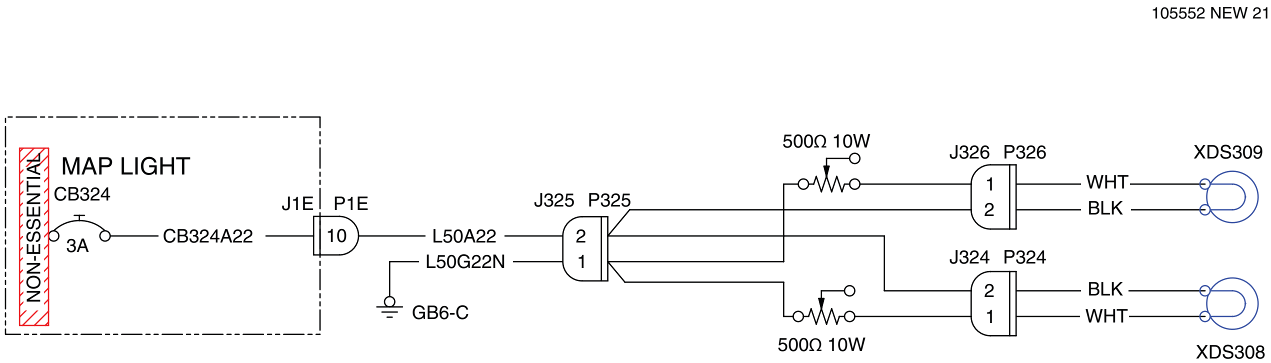
Map Lights
Figure 1 (Sheet 2 of 4)



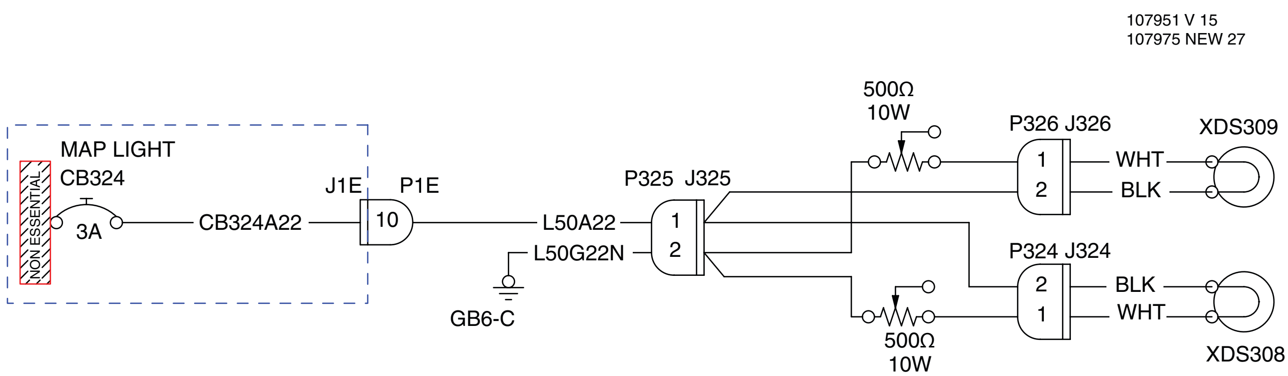




Map Lights
Figure 1 (Sheet 3 of 4)



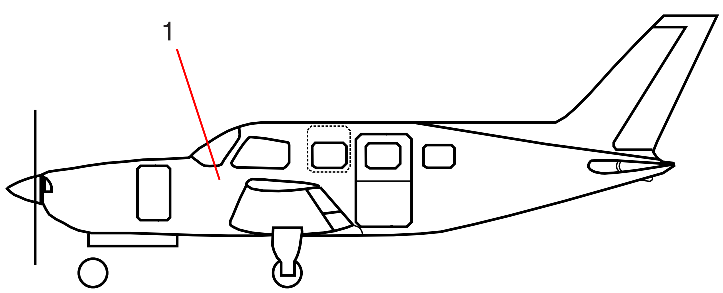
4636460, 4636463–4636480, 4636482–4636632, 4636634–4636651
4692134 and up, less 4692141 4692149 and 4692153



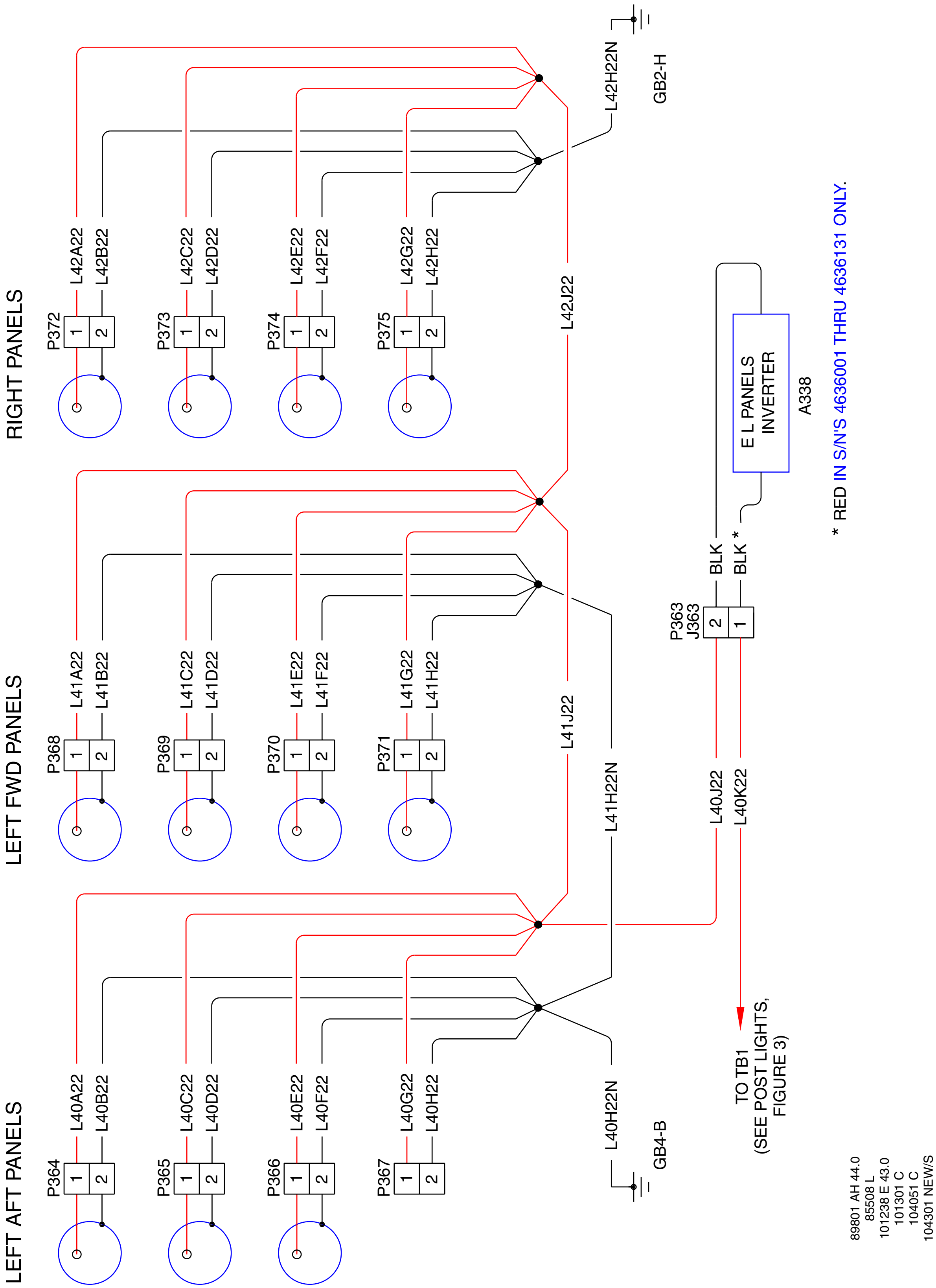
4636633, 4636652 and up

Map Lights
Figure 1 (Sheet 4 of 4)

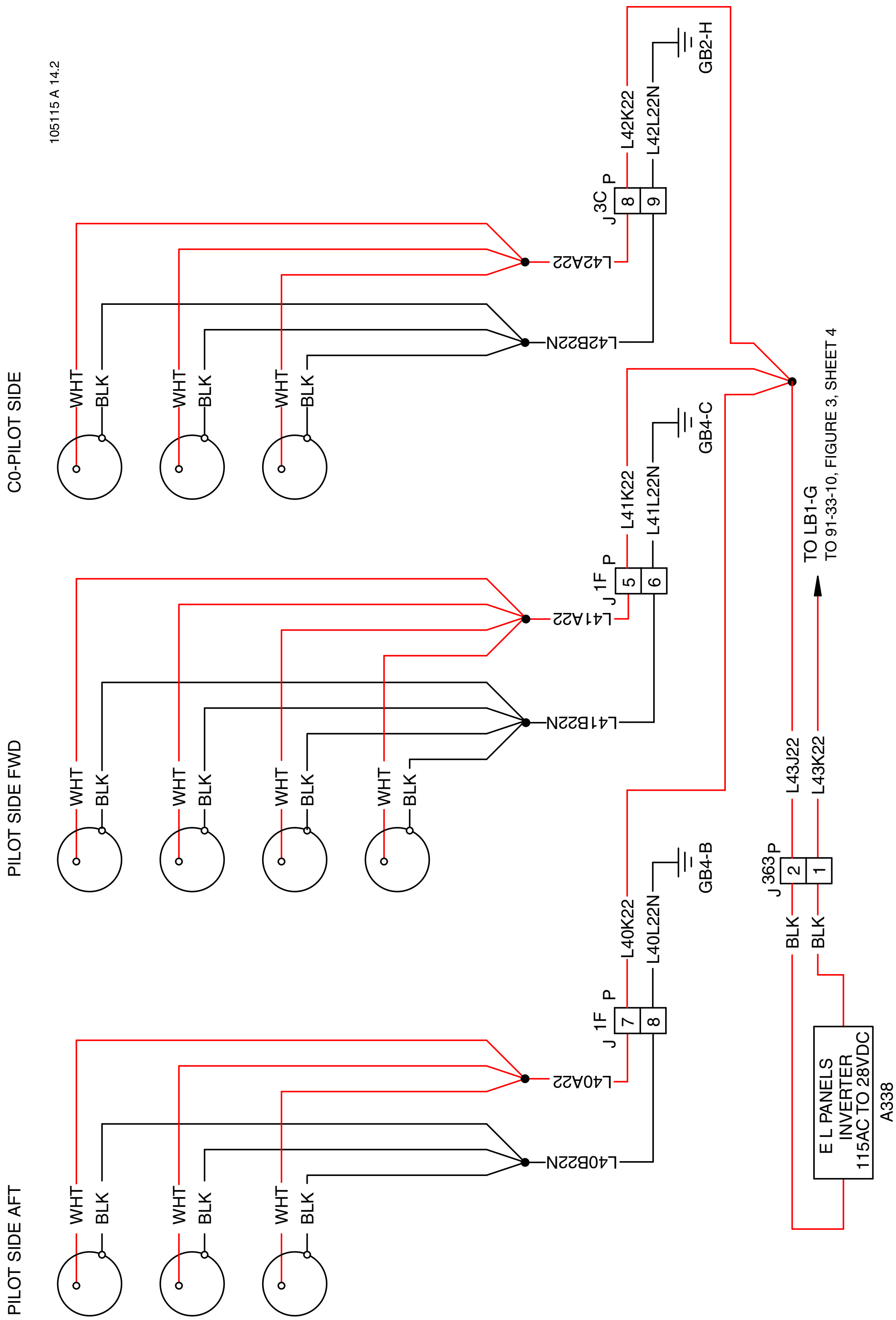
Item #	Designation	Description
1	A338	Inverter



Electroluminescent Panel Lighting (C/B Panels)
Figure 2 (Sheet 1 of 3)



Electroluminescent Panel Lighting (C/B Panels)
Figure 2 (Sheet 2 of 3)



Electroluminescent Panel Lighting (C/B Panels)
Figure 2 (Sheet 3 of 3)

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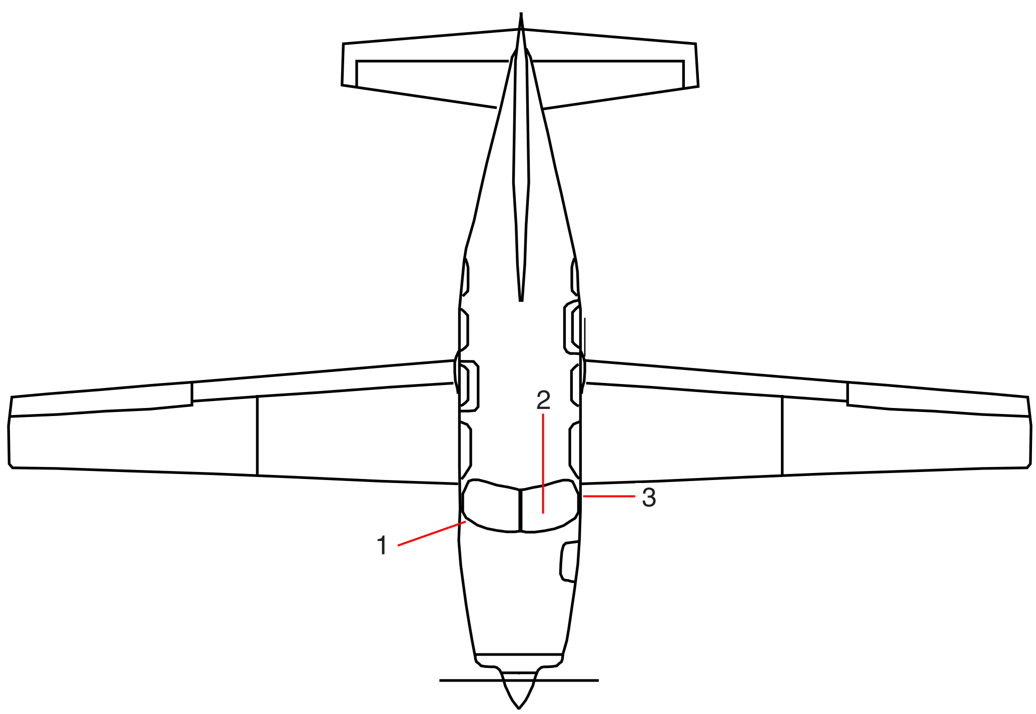
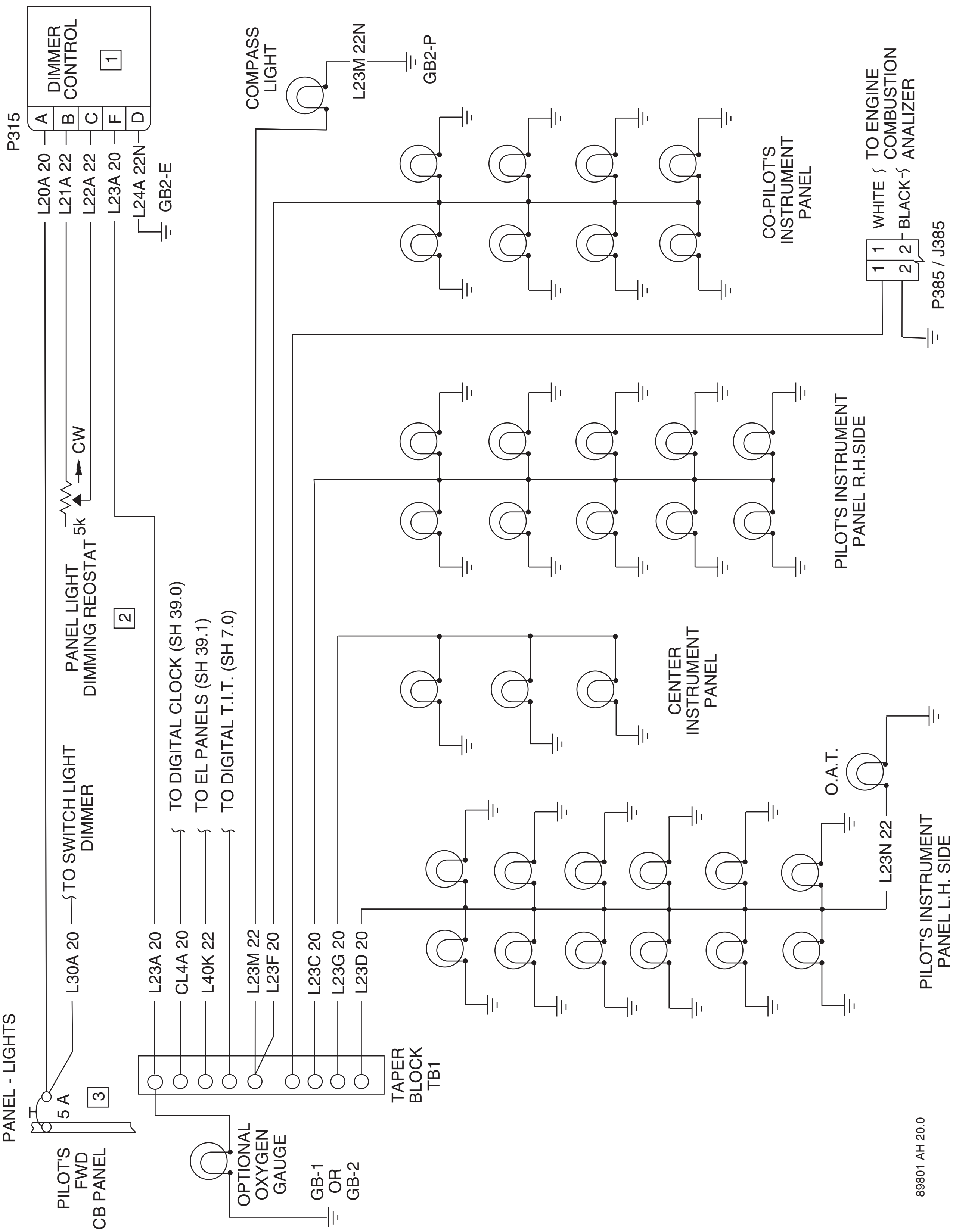
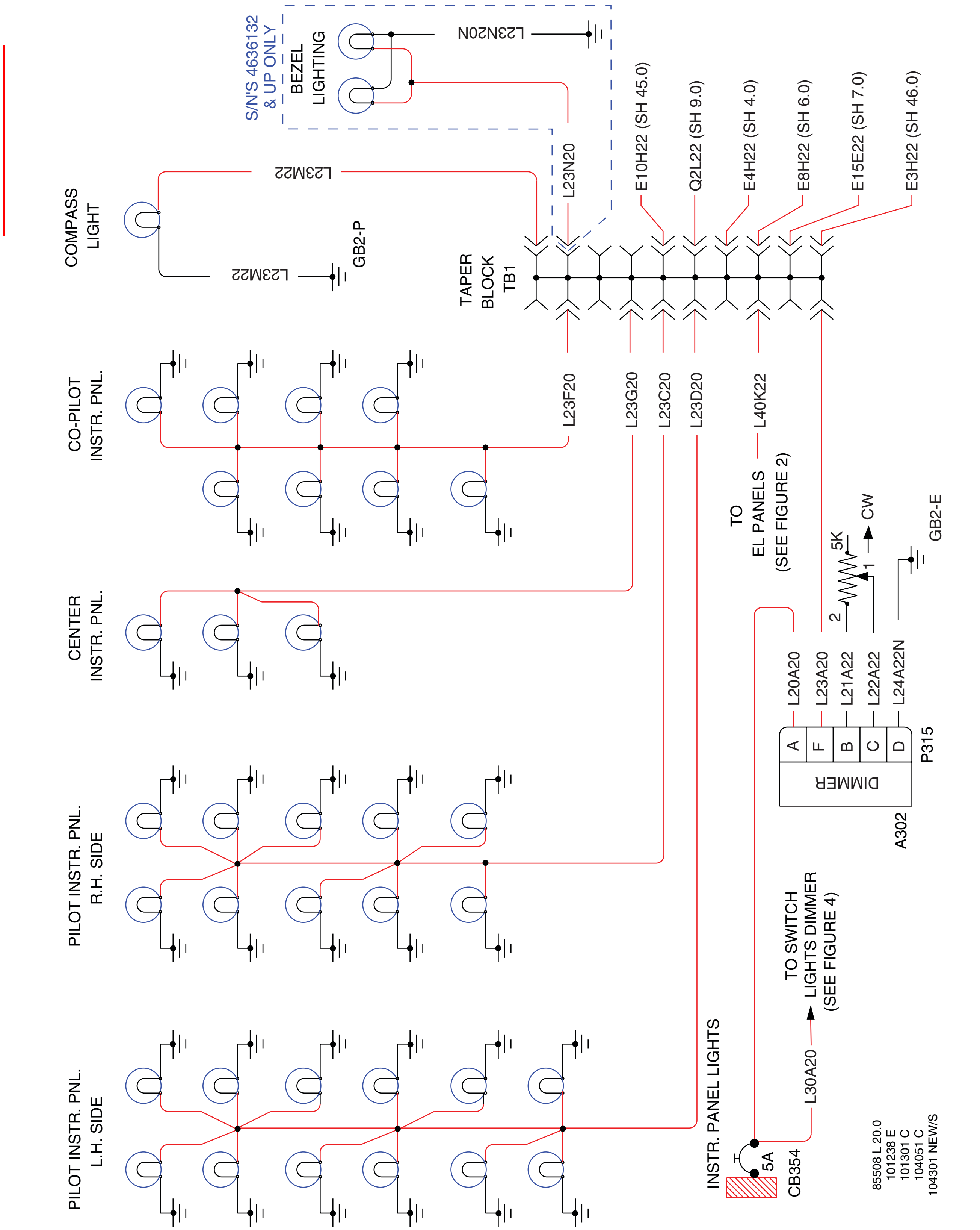
Item #	Designation	Description
1	A302	Dimmer Control
2	R301	Panel Light Dimmer Rheostat
3	CB354	Circuit Breaker - Instr. Panel Lights (5 Amp)
		Post Light Assembly



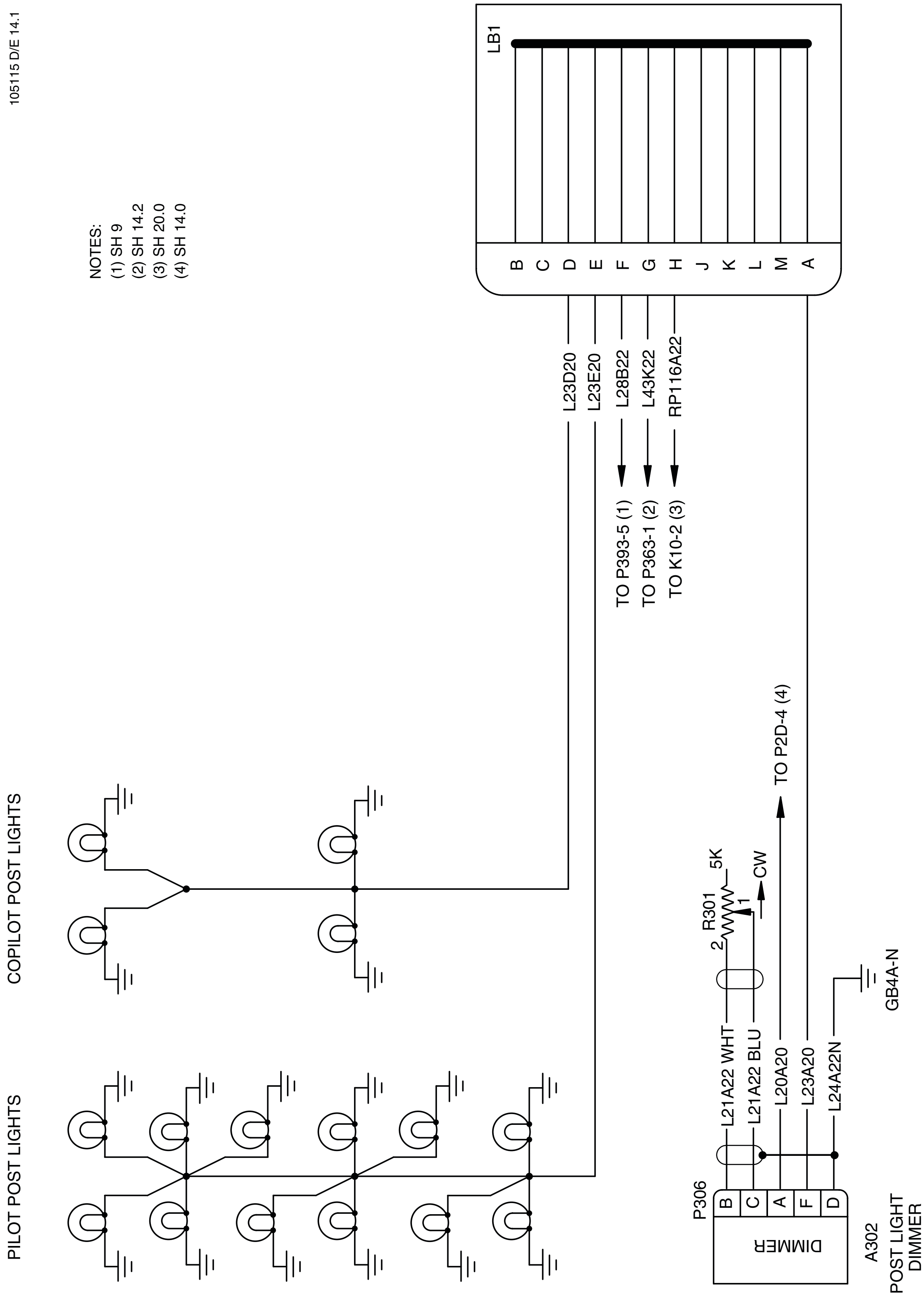
Instrument Panel - Post Lights / Dimmer
Figure 3 (Sheet 1 of 7)



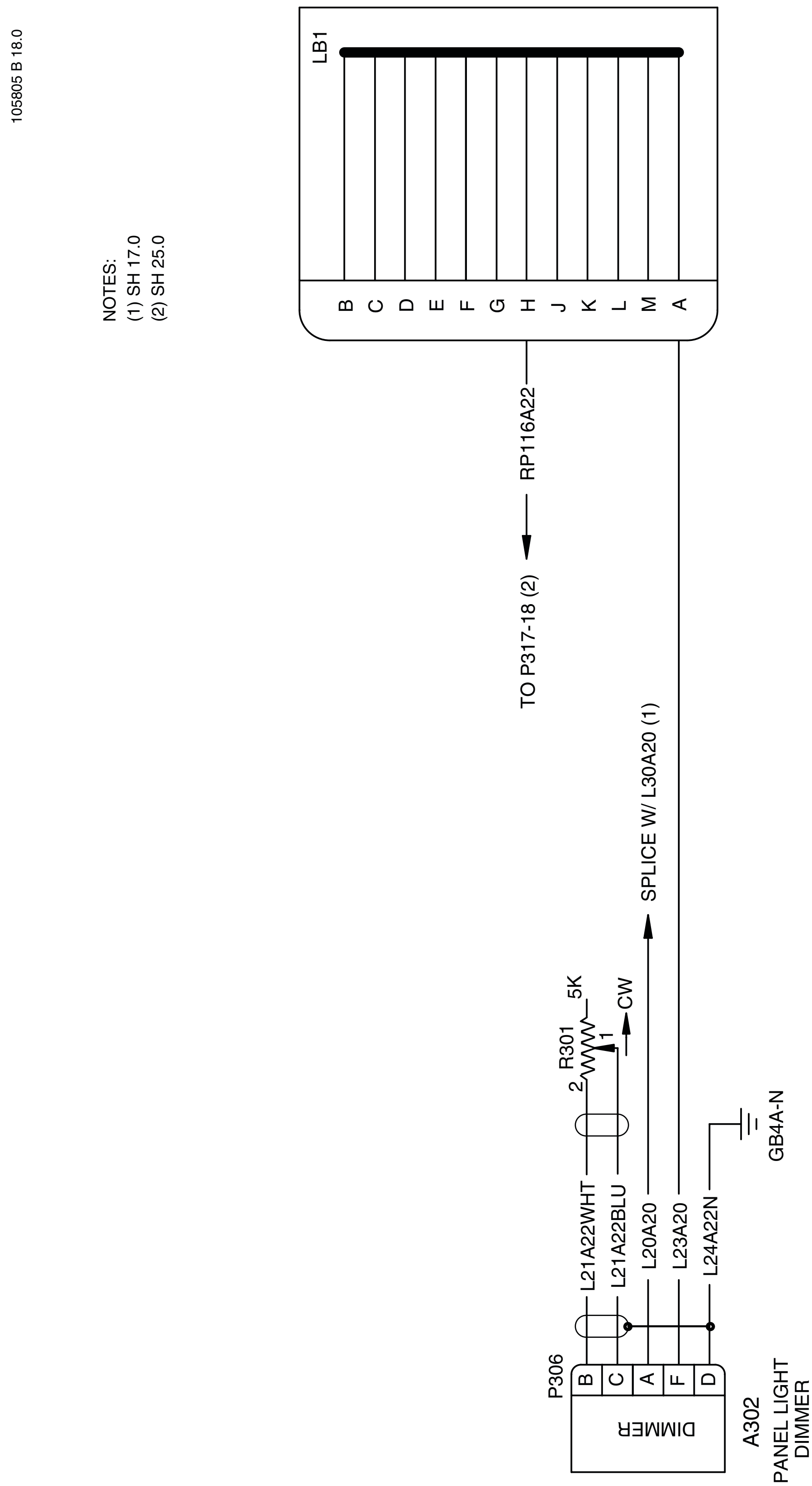
Instrument Panel - Post Lights / Dimmer
Figure 3 (Sheet 2 of 7)



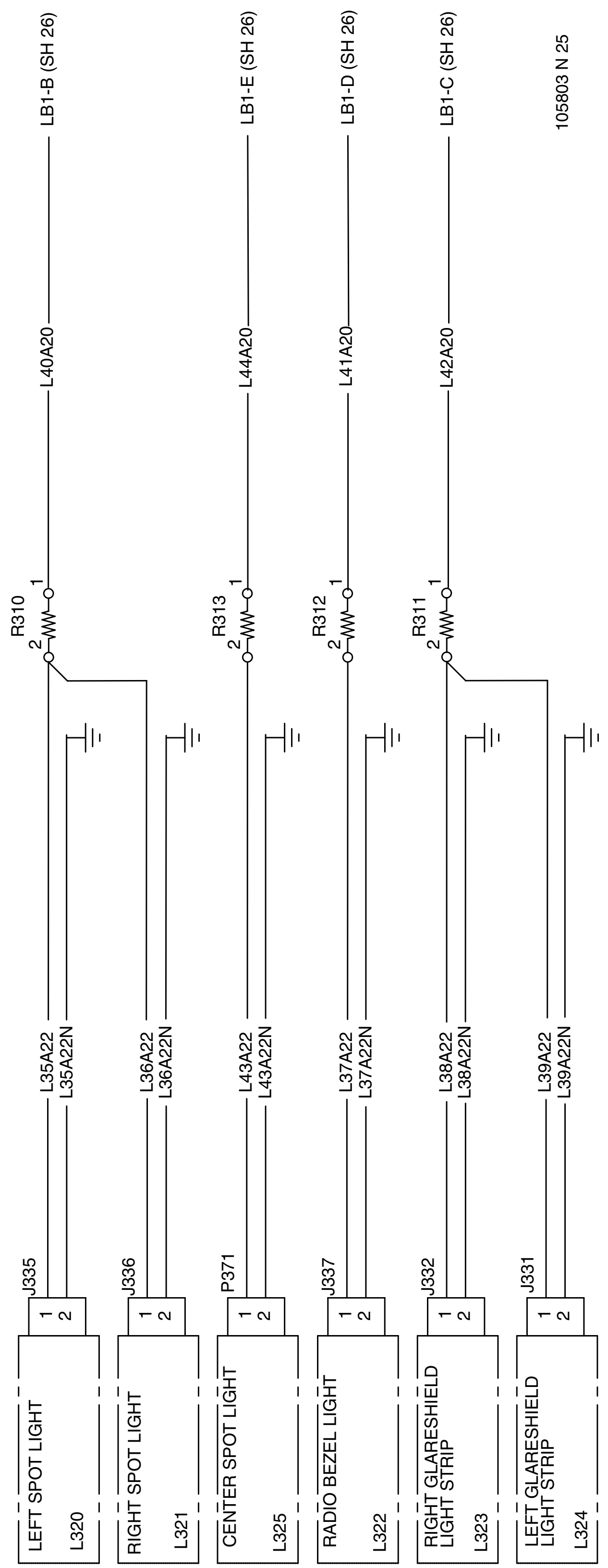
Instrument Panel - Post Lights / Dimmer
Figure 3 (Sheet 3 of 7)



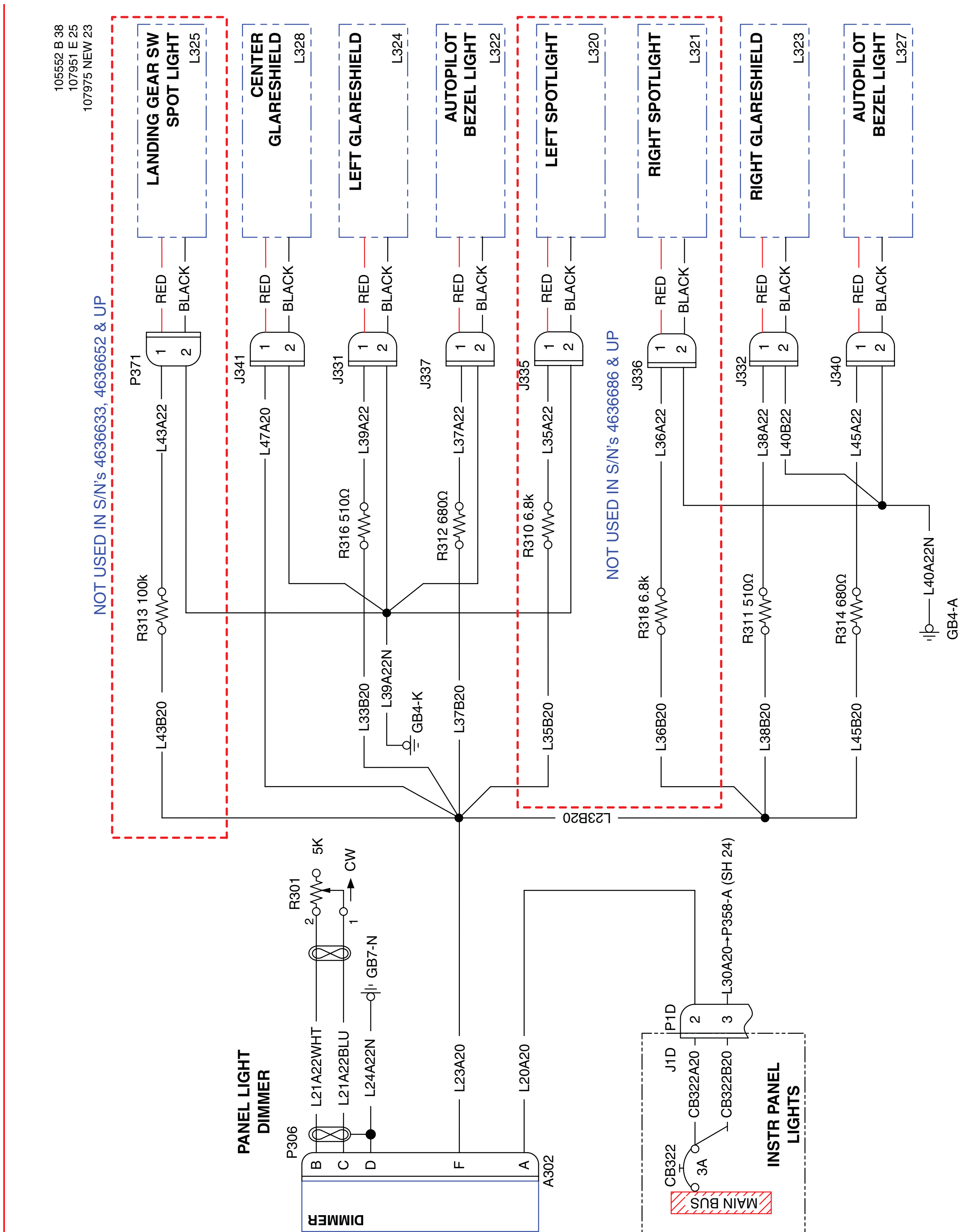
Instrument Panel - Post Lights / Dimmer
Figure 3 (Sheet 4 of 7)



Instrument Panel - Post Lights / Dimmer
Figure 3 (Sheet 5 of 7)



Instrument Panel - Post Lights / Dimmer
Figure 3 (Sheet 6 of 7)



Instrument Panel - Post Lights / Dimmer
Figure 3 (Sheet 7 of 7)

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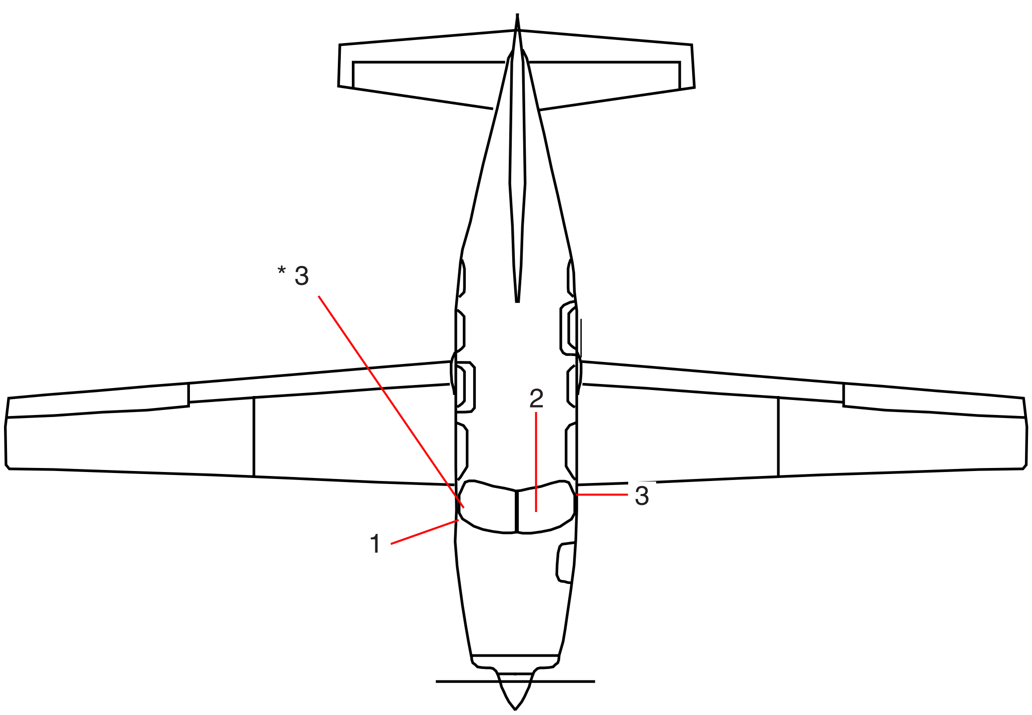
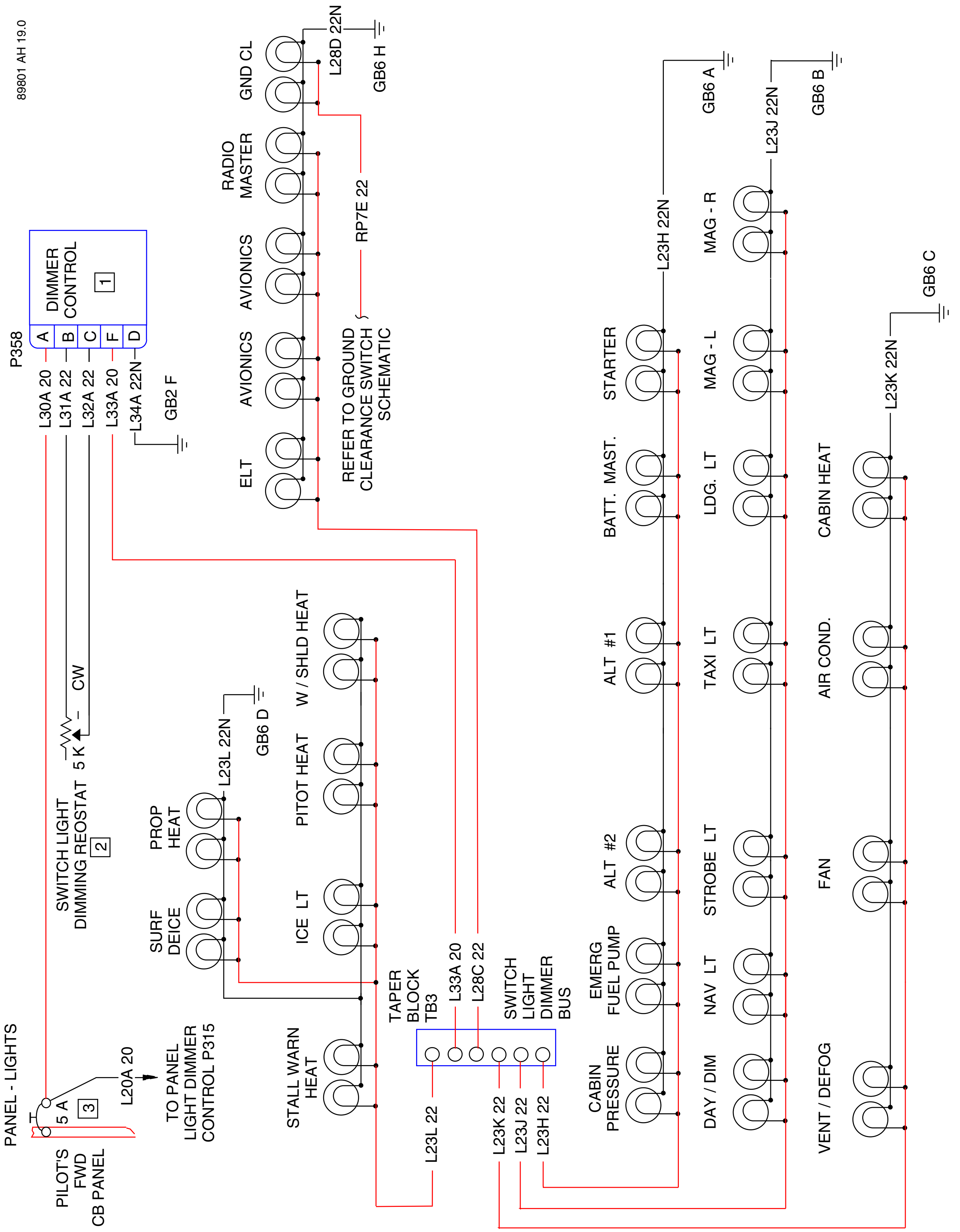
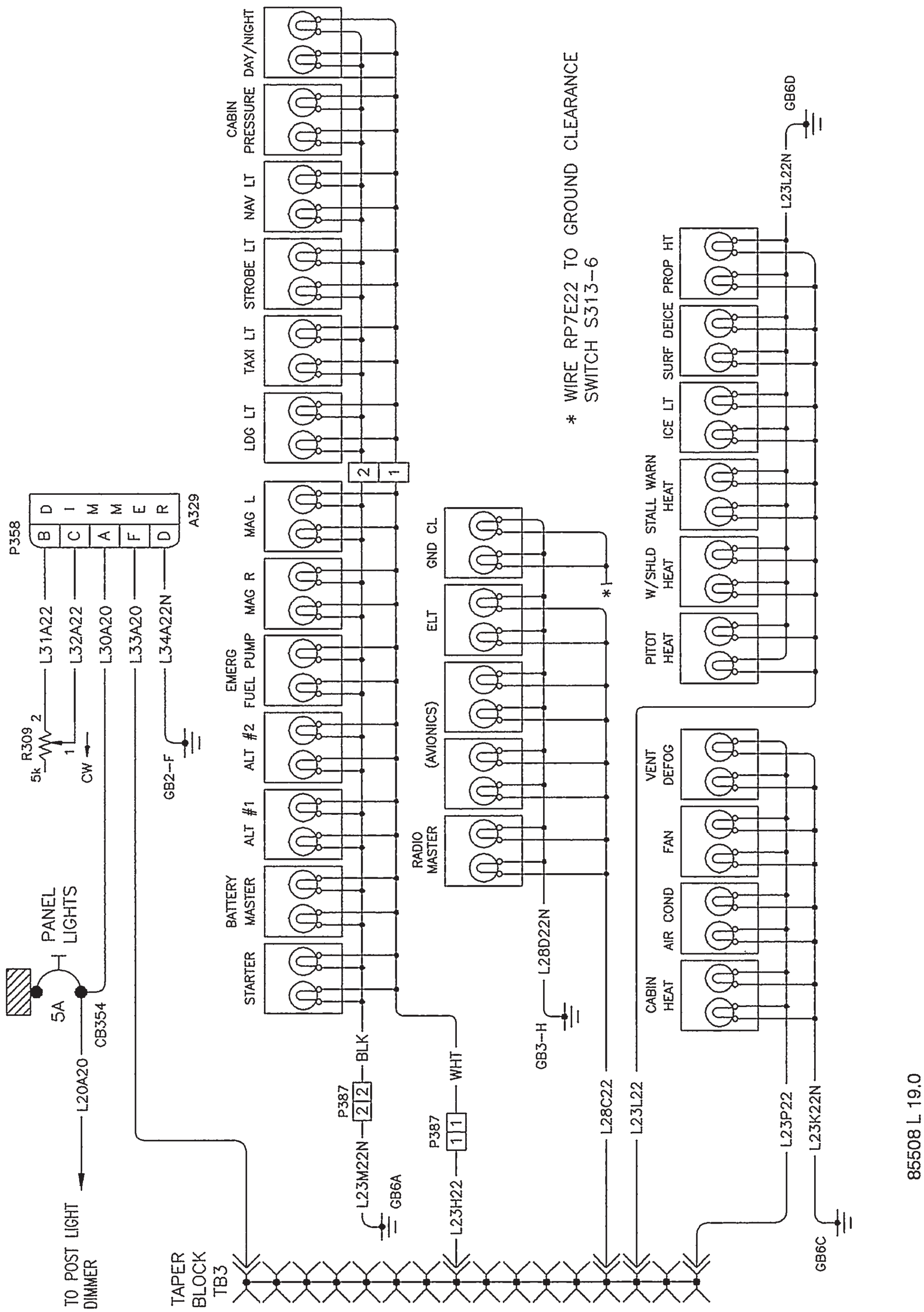
Item #	Designation	Description
1	A329	Dimmer Control
2	R309	Panel Light Dimmer Rheostat
3	CB354, CB21*	Circuit Breaker - Instr. Panel Lights (5 Amp)



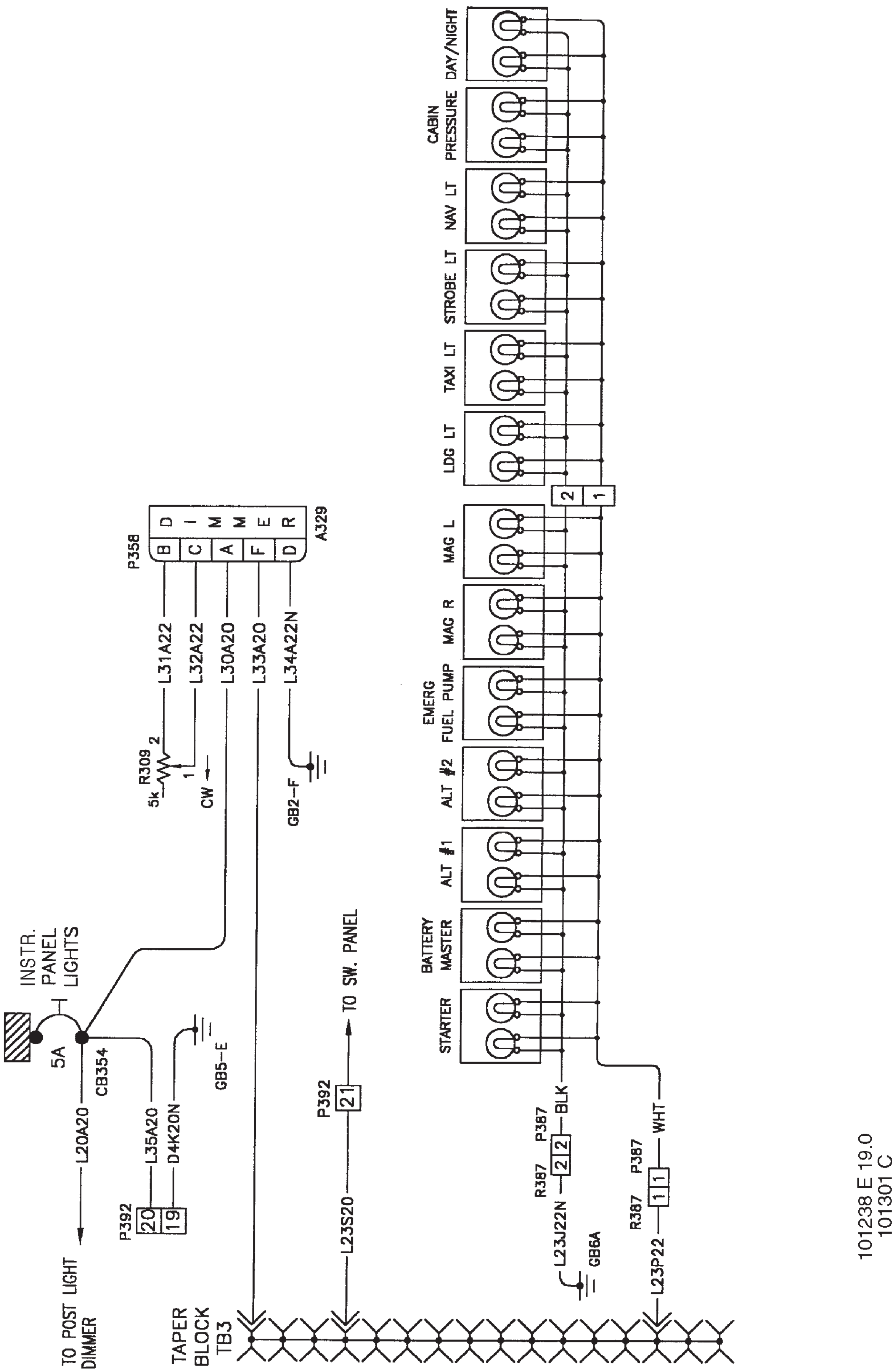
Instrument Panel Switch Lights
Figure 4 (Sheet 1 of 11)



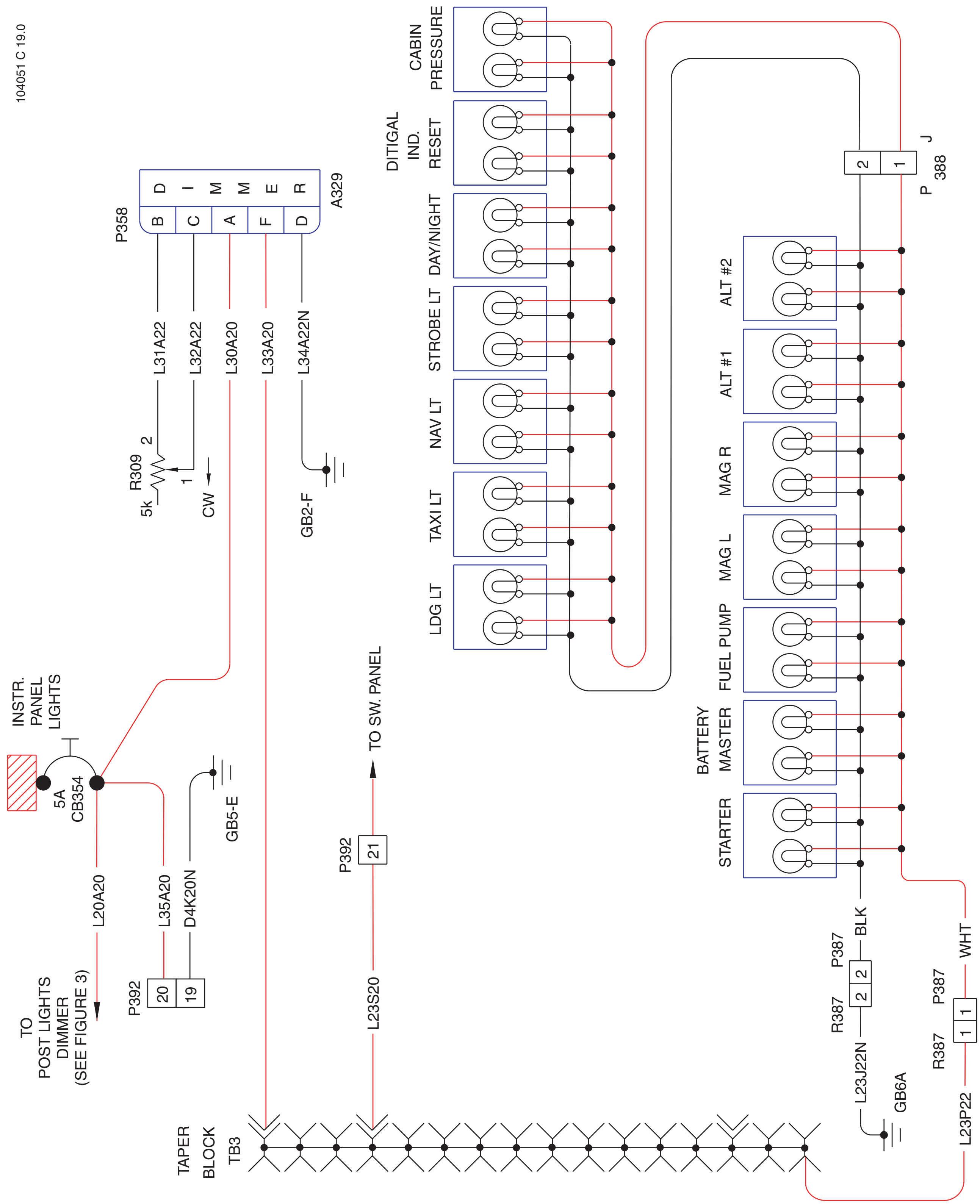
Instrument Panel Switch Lights
Figure 4 (Sheet 2 of 11)



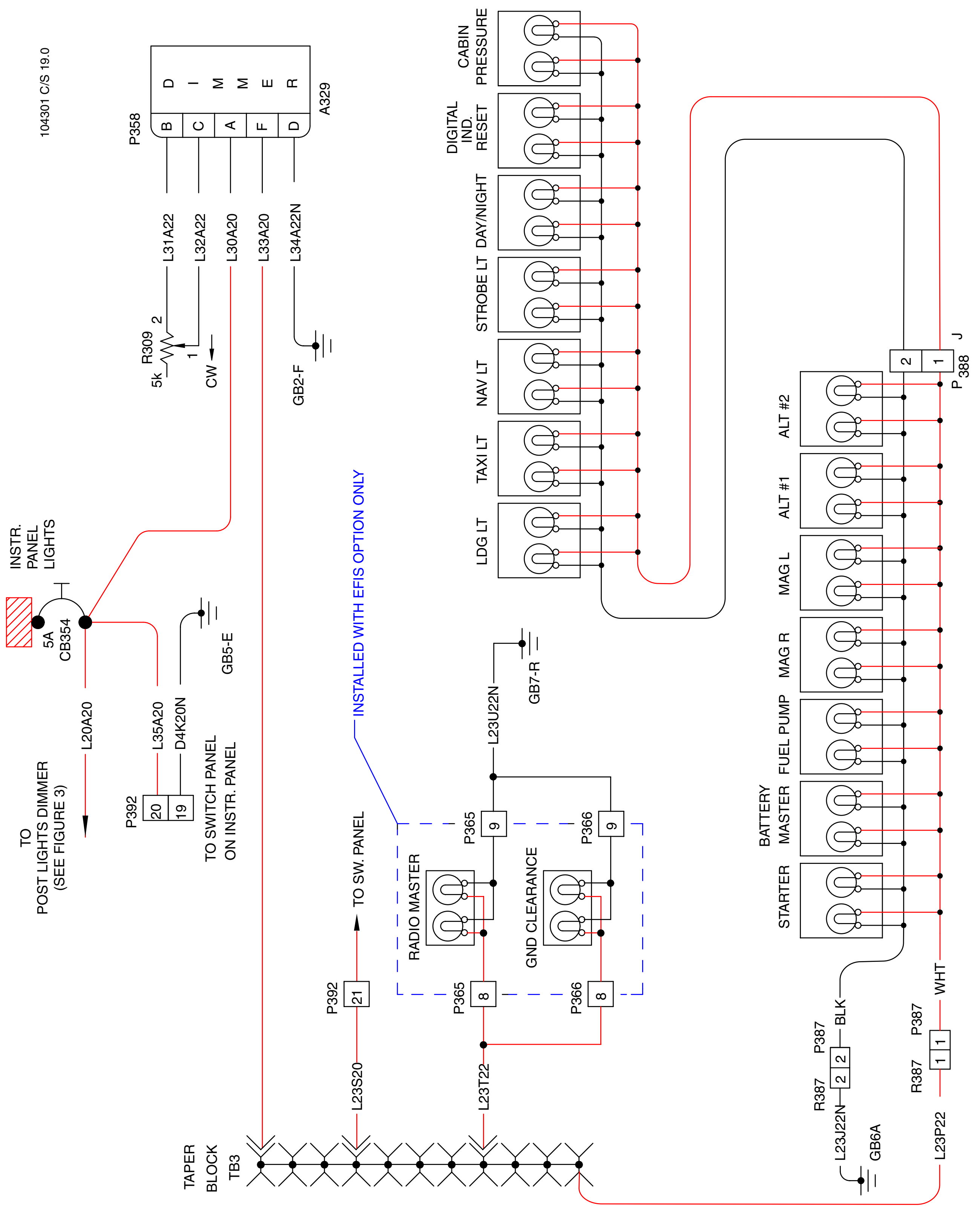
Instrument Panel Switch Lights
Figure 4 (Sheet 3 of 11)



Instrument Panel Switch Lights
Figure 4 (Sheet 4 of 11)



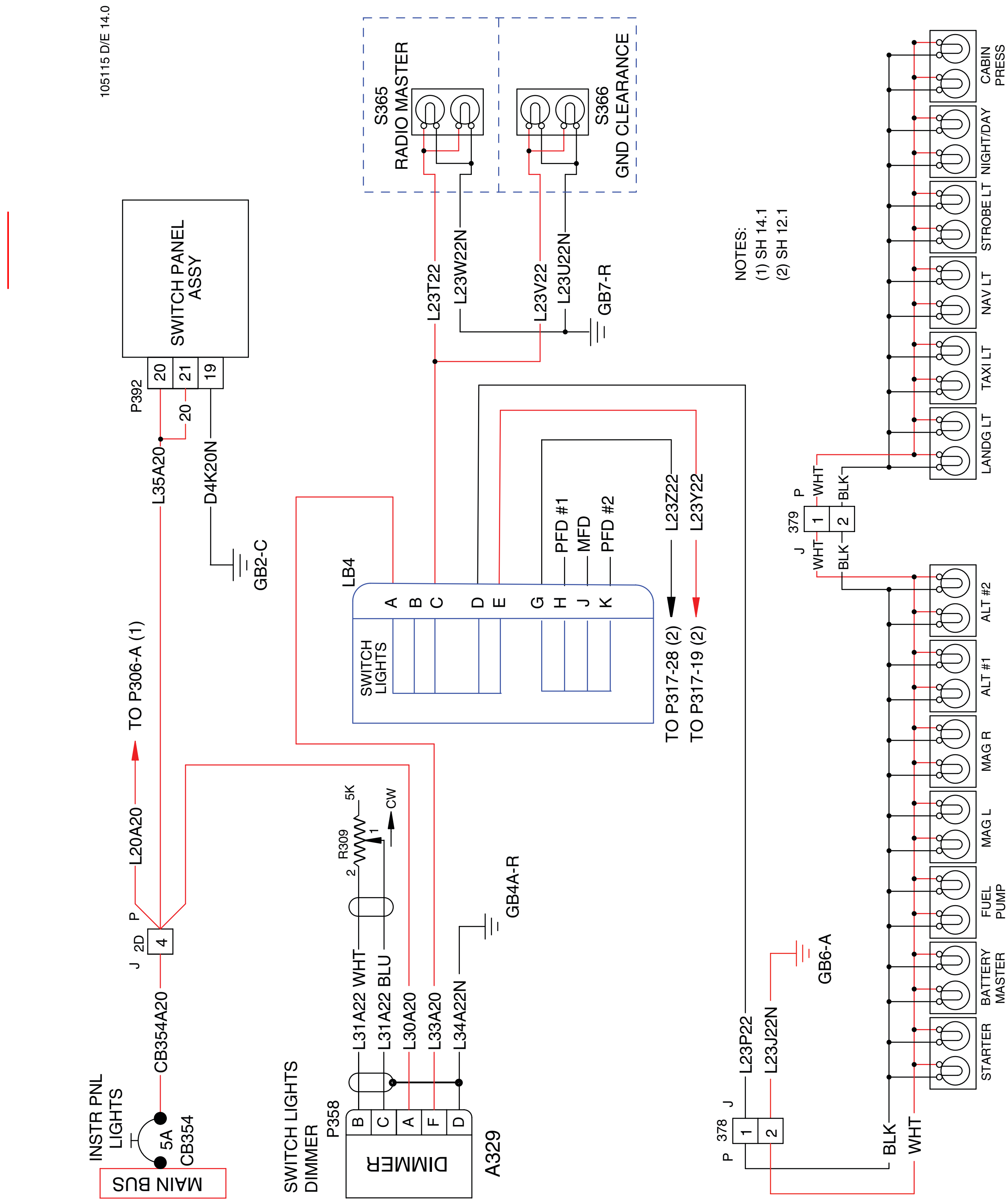
Instrument Panel Switch Lights
Figure 4 (Sheet 5 of 11)



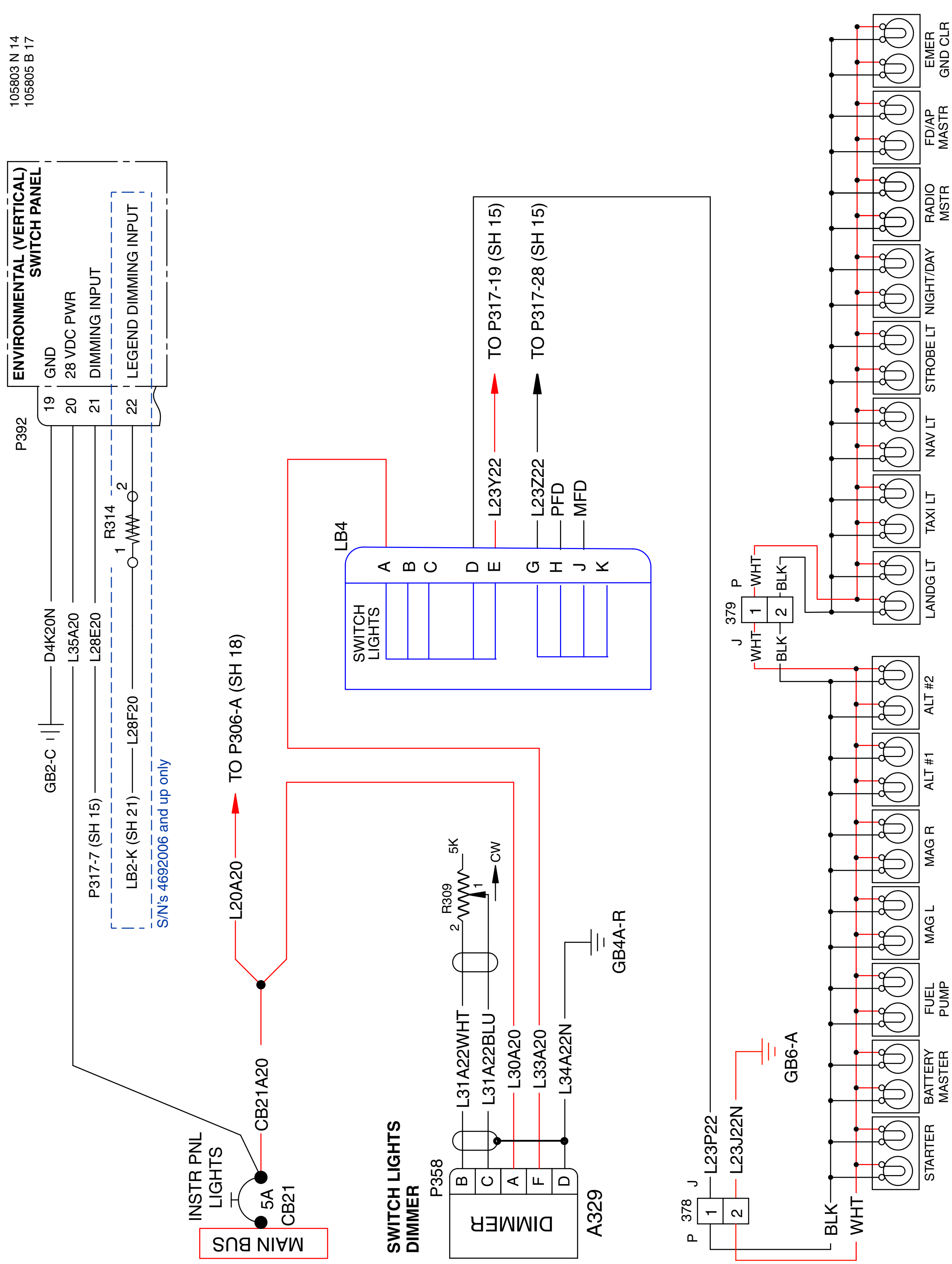
Instrument Panel Switch Lights
Figure 4 (Sheet 6 of 11)

Effectivity
4636299, 4636314–4636374

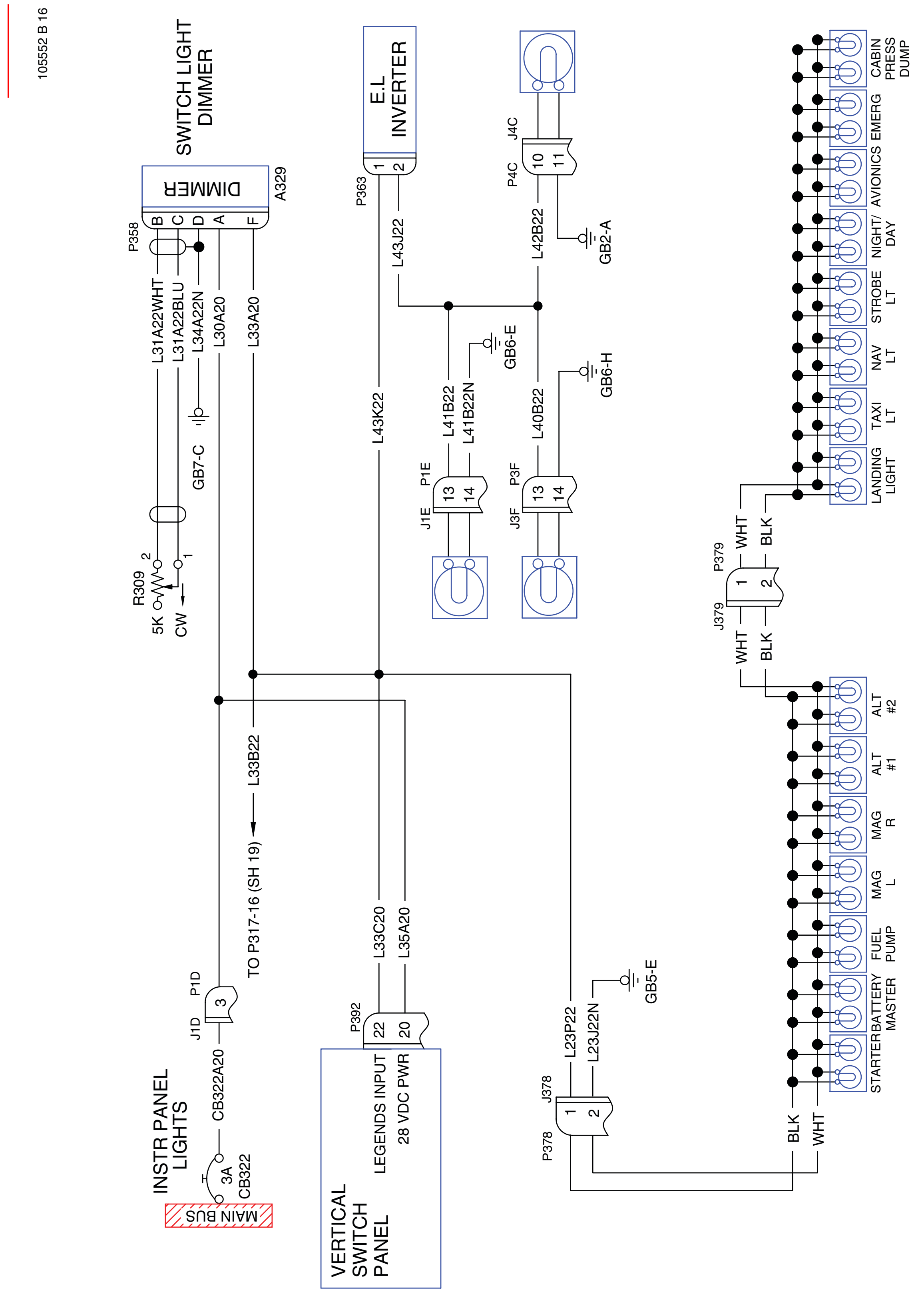
91331022



Instrument Panel Switch Lights
Figure 4 (Sheet 7 of 11)

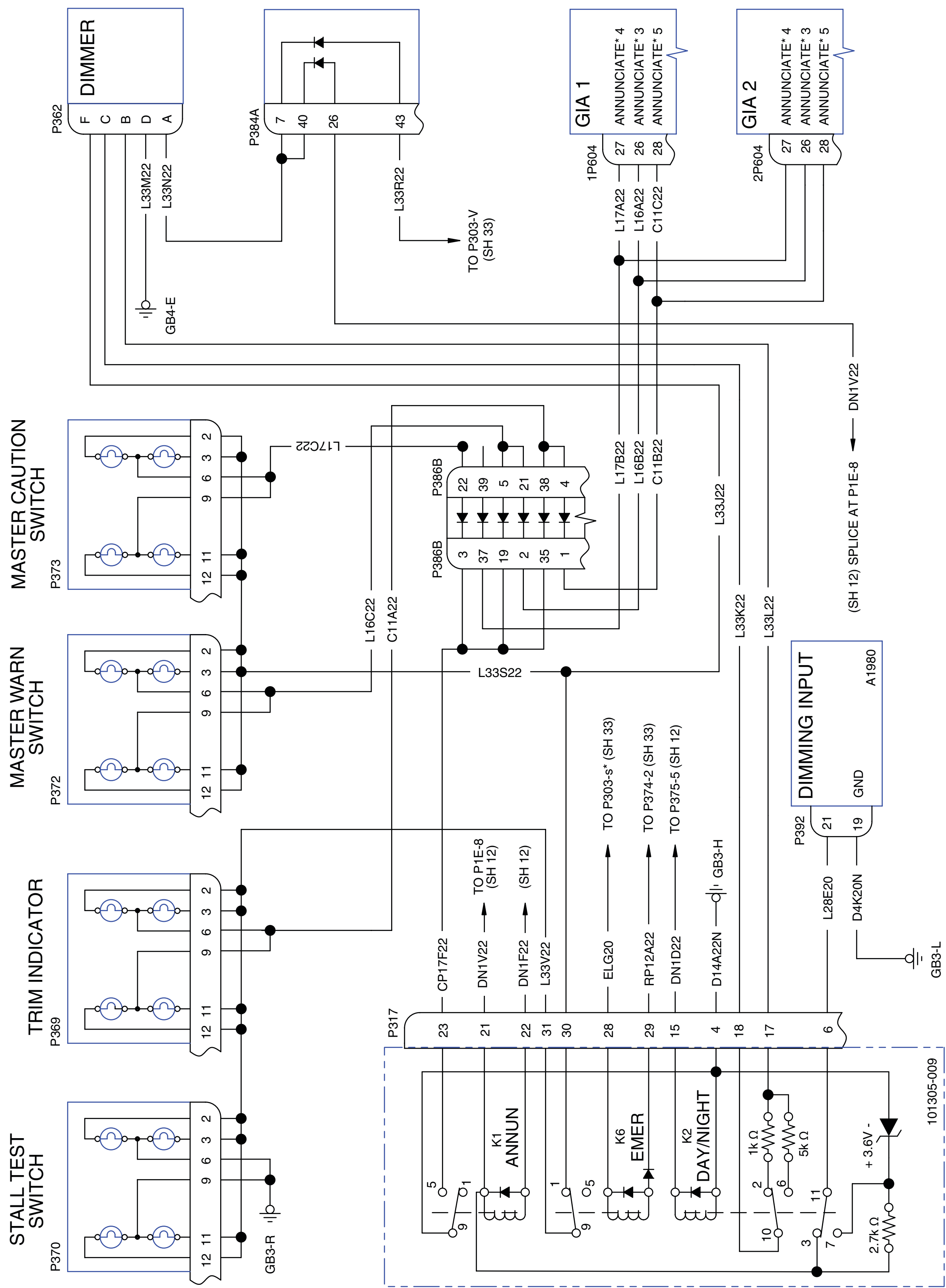


Instrument Panel Switch Lights
Figure 4 (Sheet 8 of 11)



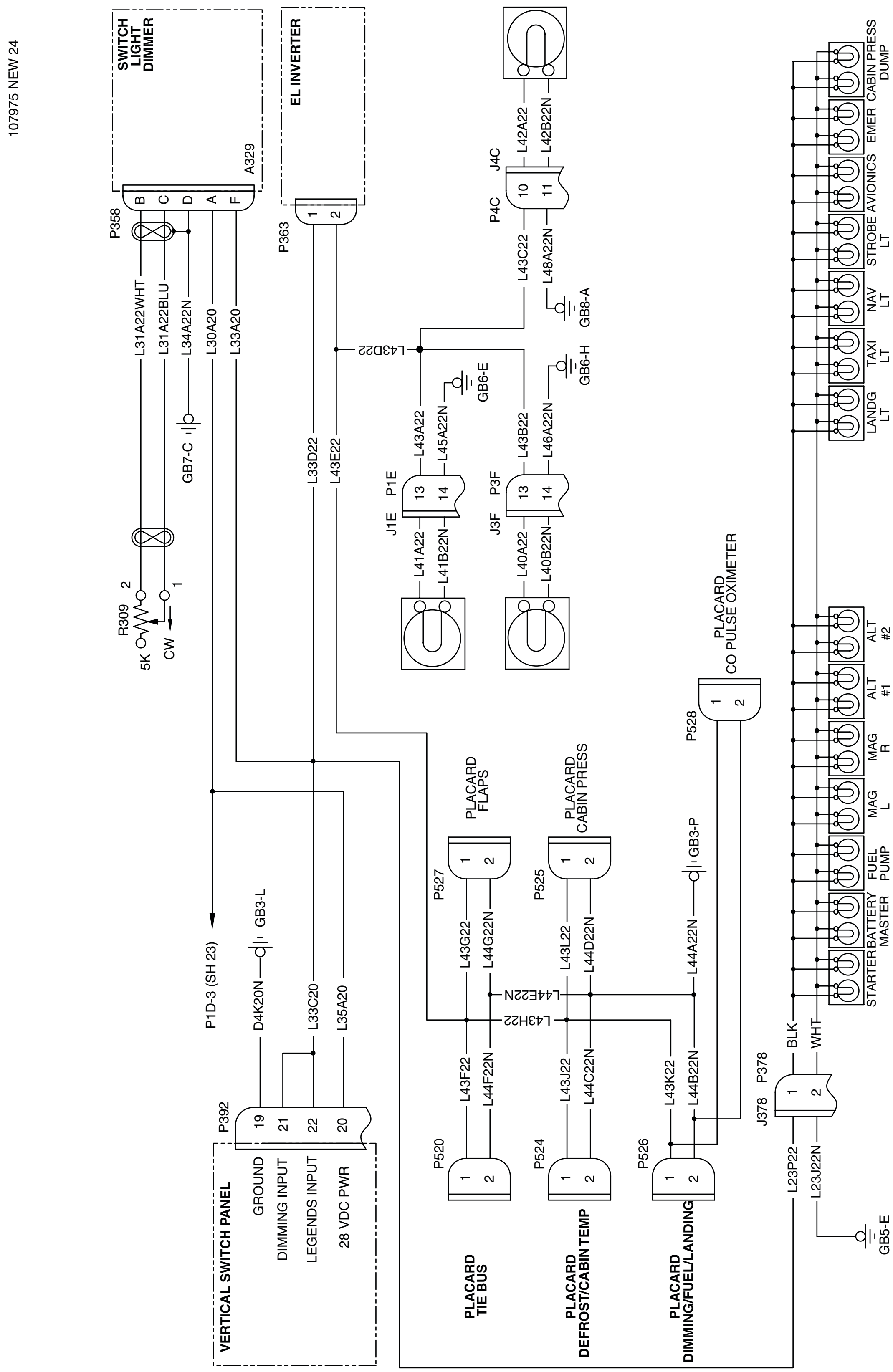
Instrument Panel Switch Lights
Figure 4 (Sheet 9 of 11)

105552 B 40



Instrument Panel Switch Lights
Figure 4 (Sheet 10 of 11)

Effectivity
4636460, 4636463–4636651, less 4636481 and 4636633
4692134 and up, less 4692141, 4692149, 4692153



Instrument Panel Switch Lights
Figure 4 (Sheet 11 of 11)

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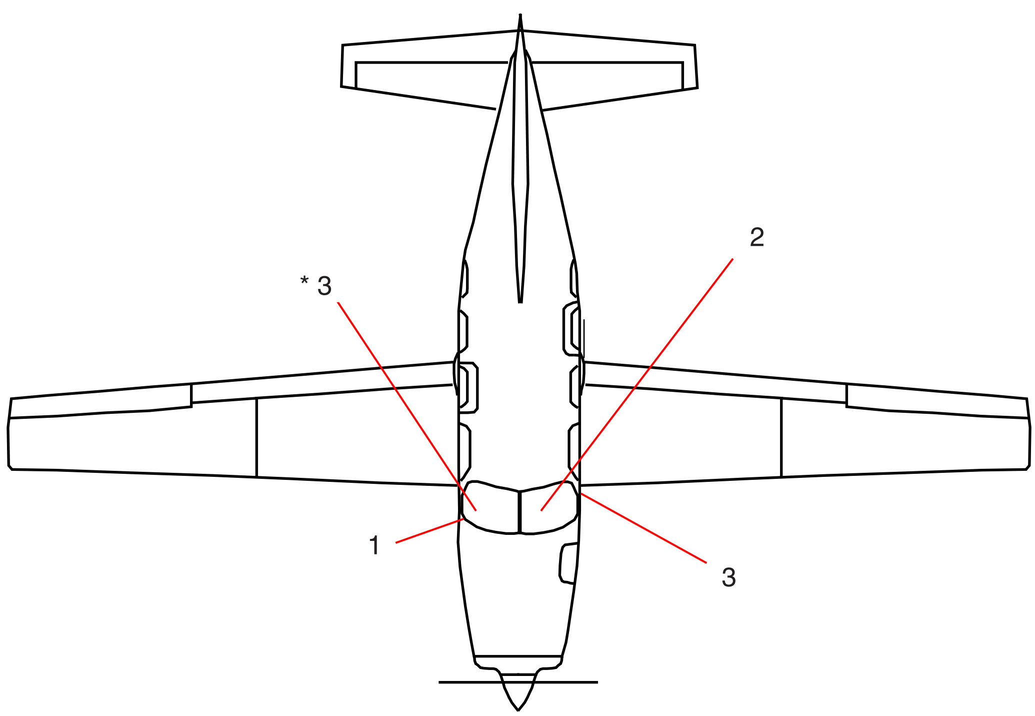
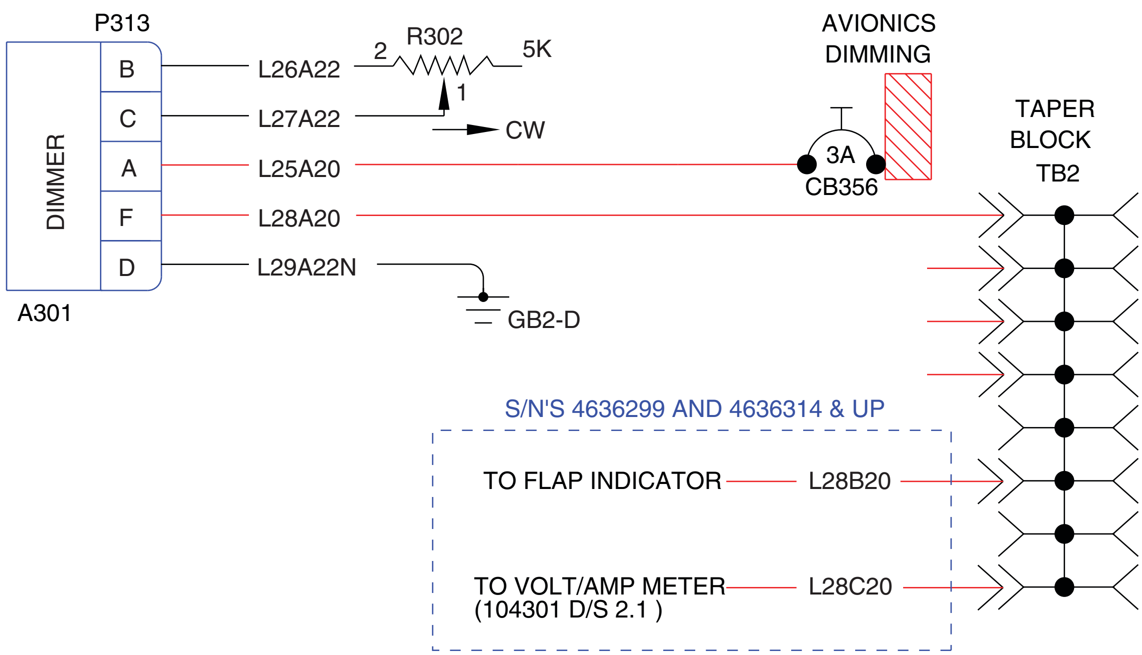
Item #	Designation	Description
1	A301	Dimmer Control
2	R302	Panel Light Dimmer Rheostat
3	CB356, CB22*	Circuit Breaker - Avionics Dimming (3 Amp)

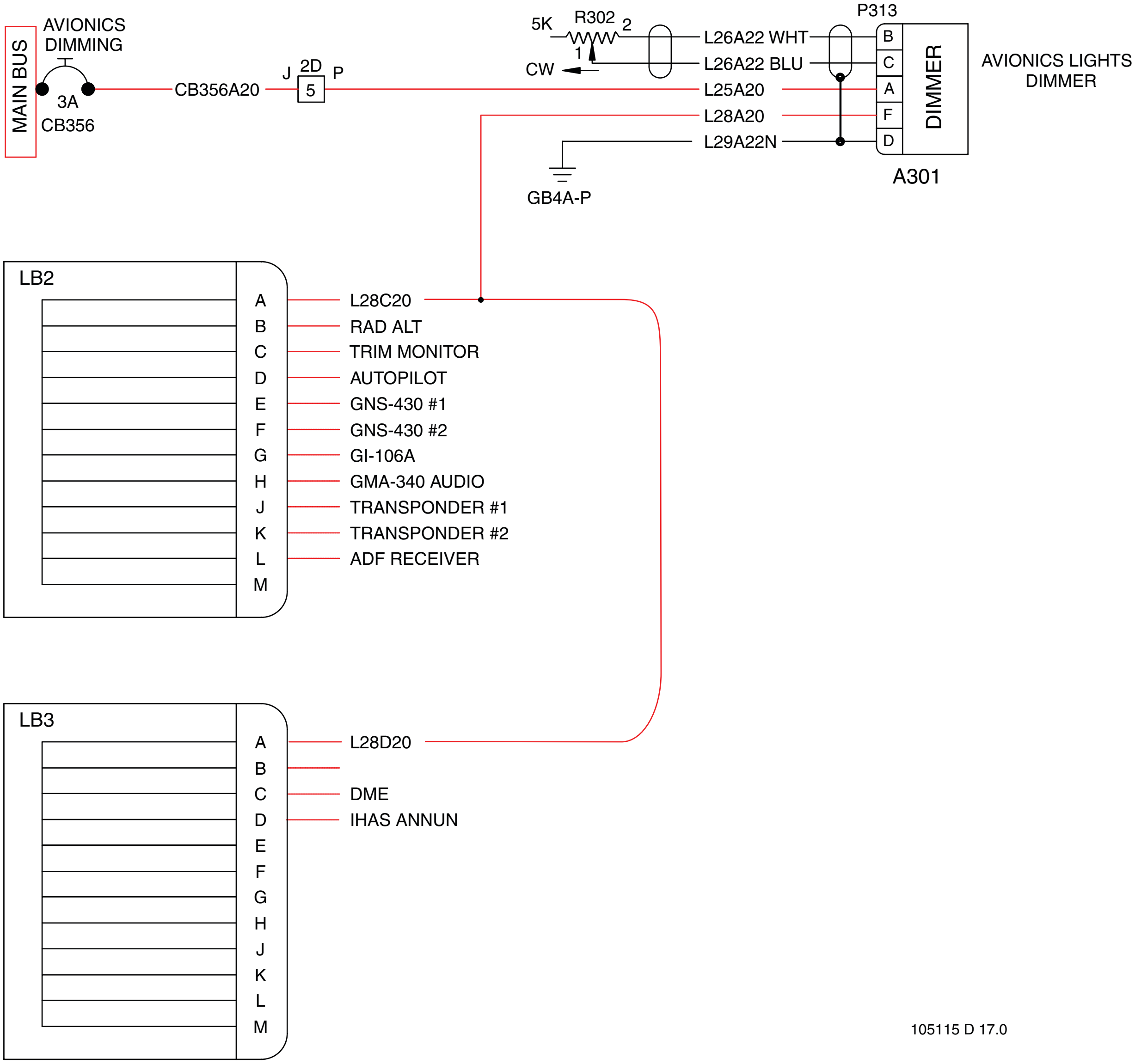


Avionics Lighting
Figure 5 (Sheet 1 of 7)

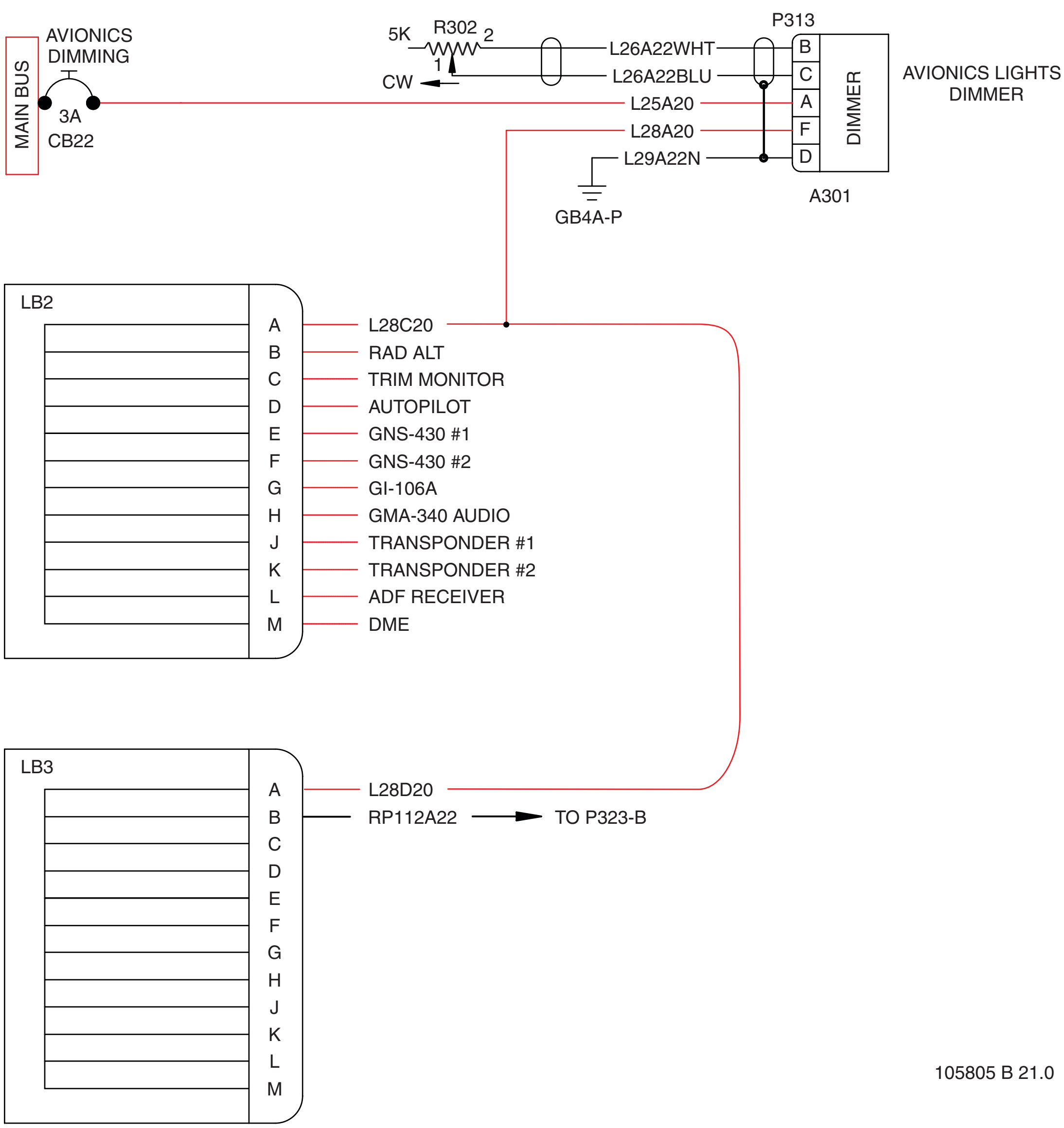


89801 AH 36.0
85508 L
101238 E
101301 C
104051 C
104301 NEW/S

Avionics Lighting
Figure 5 (Sheet 2 of 7)

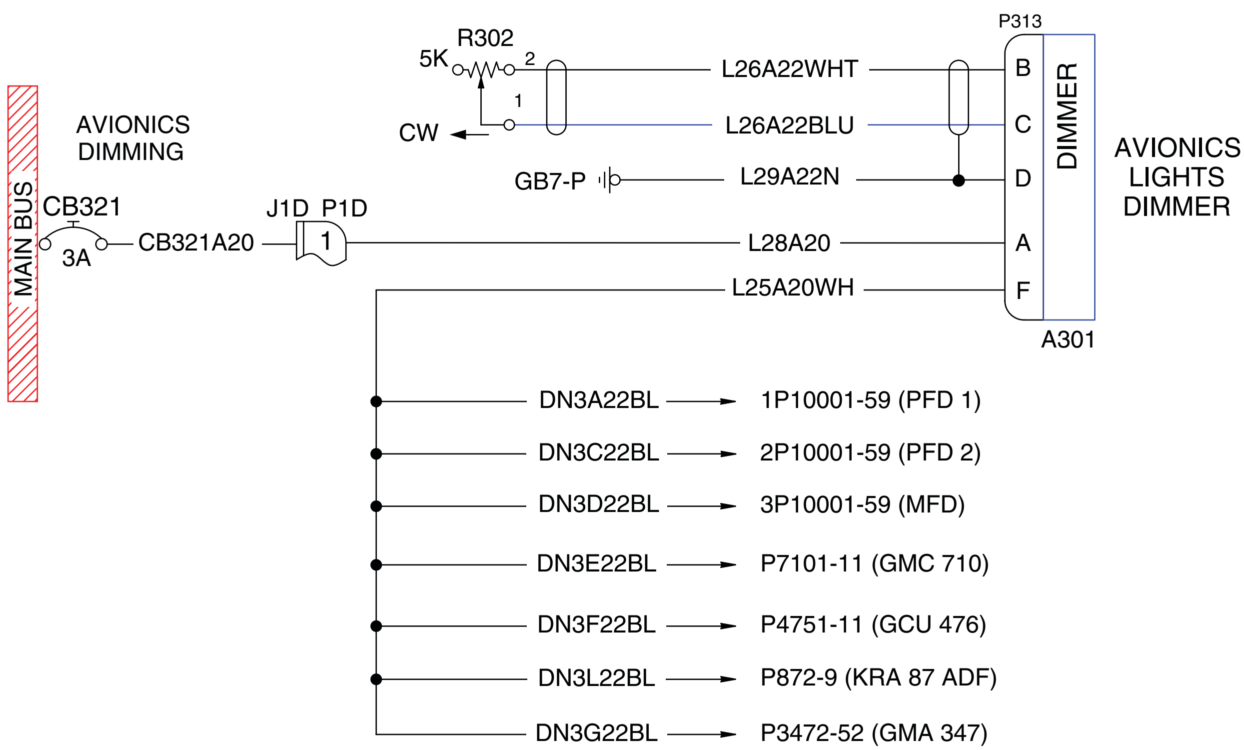


Avionics Lighting
Figure 5 (Sheet 3 of 7)

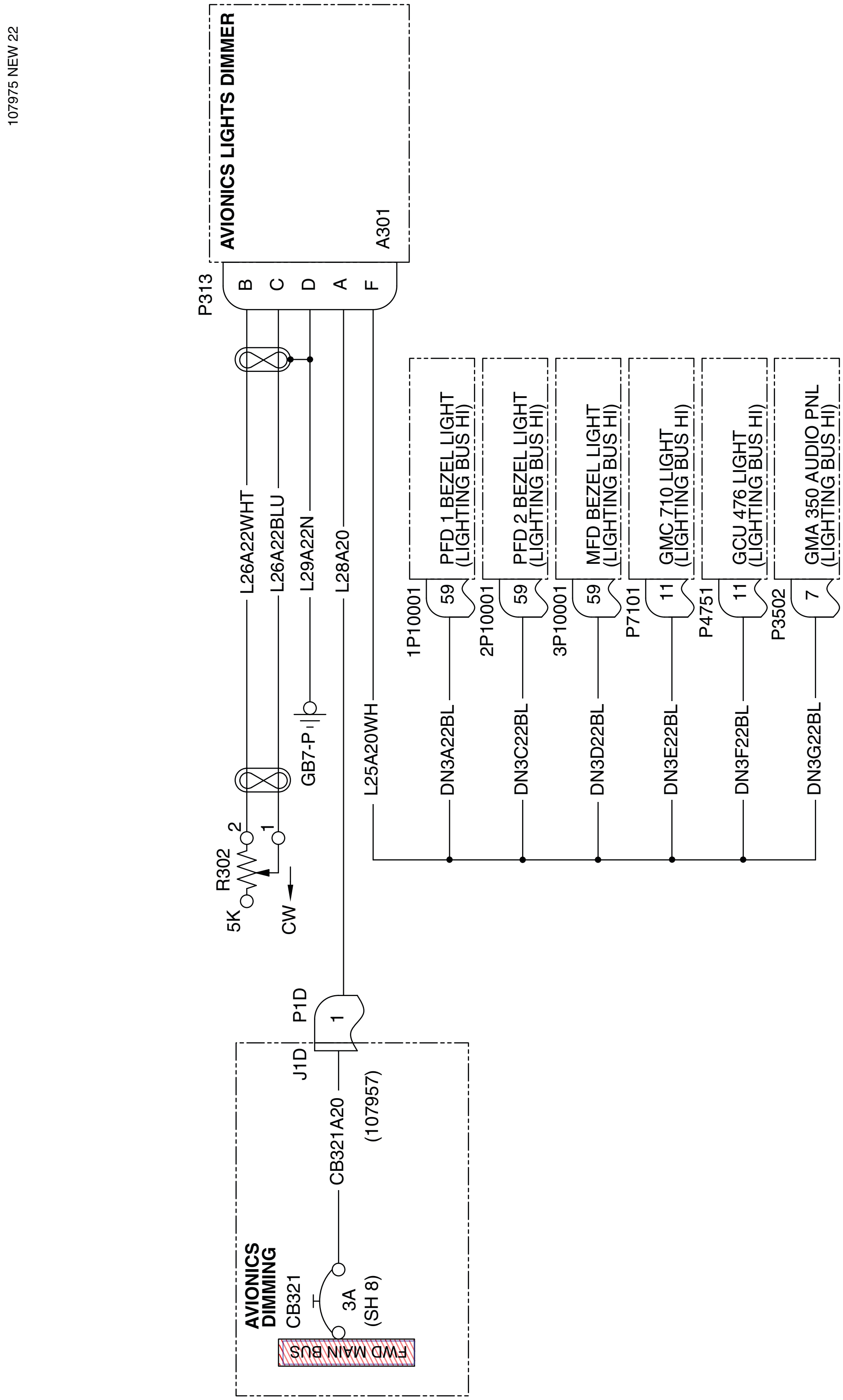


Avionics Lighting
Figure 5 (Sheet 4 of 7)

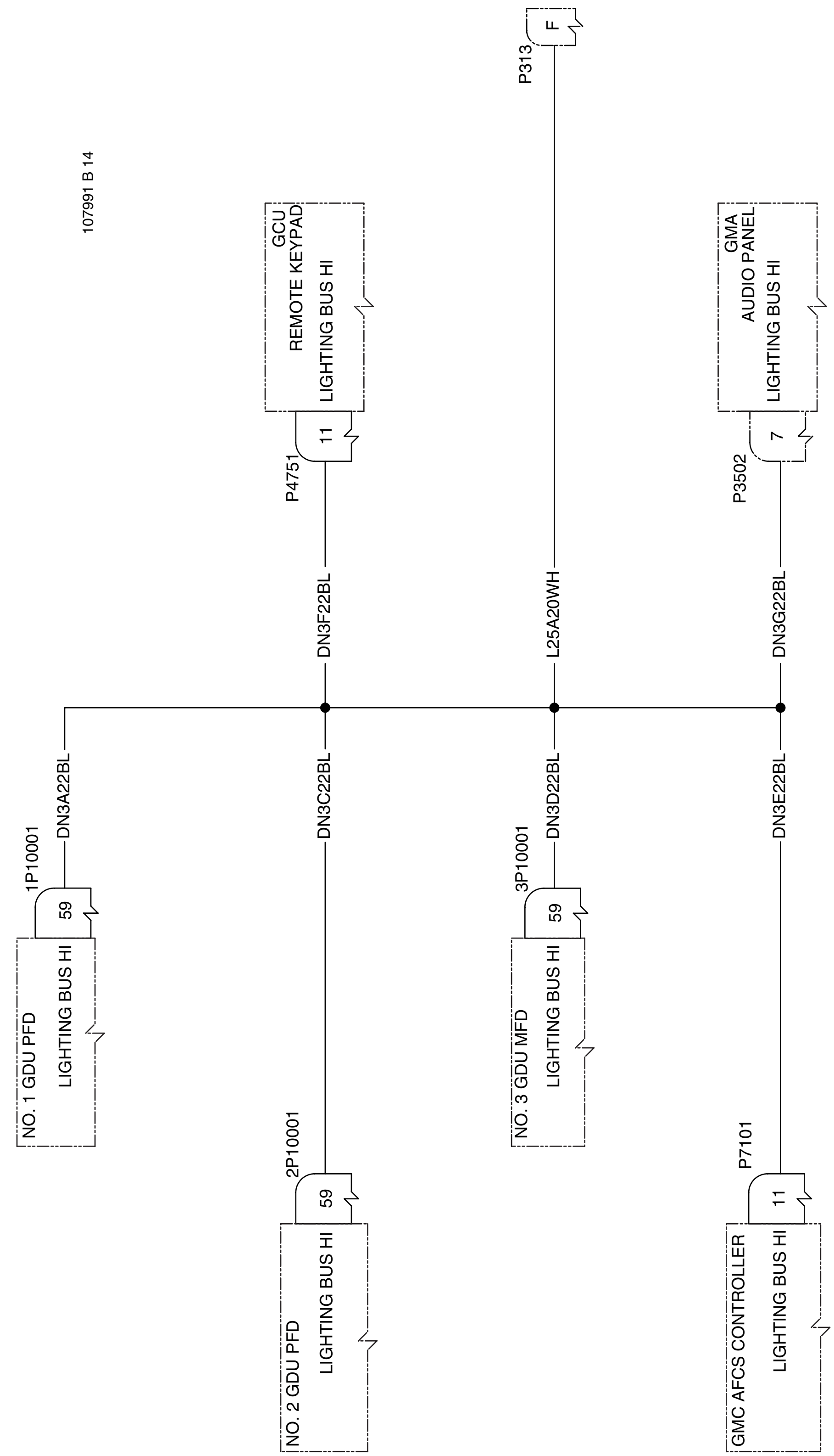
105552 B 36
105553 G 18



Avionics Lighting
Figure 5 (Sheet 5 of 7)



Avionics Lighting
Figure 5 (Sheet 6 of 7)



Avionics Lighting
Figure 5 (Sheet 7 of 7)

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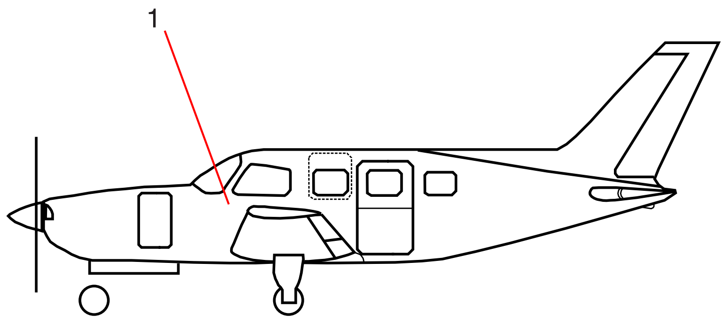
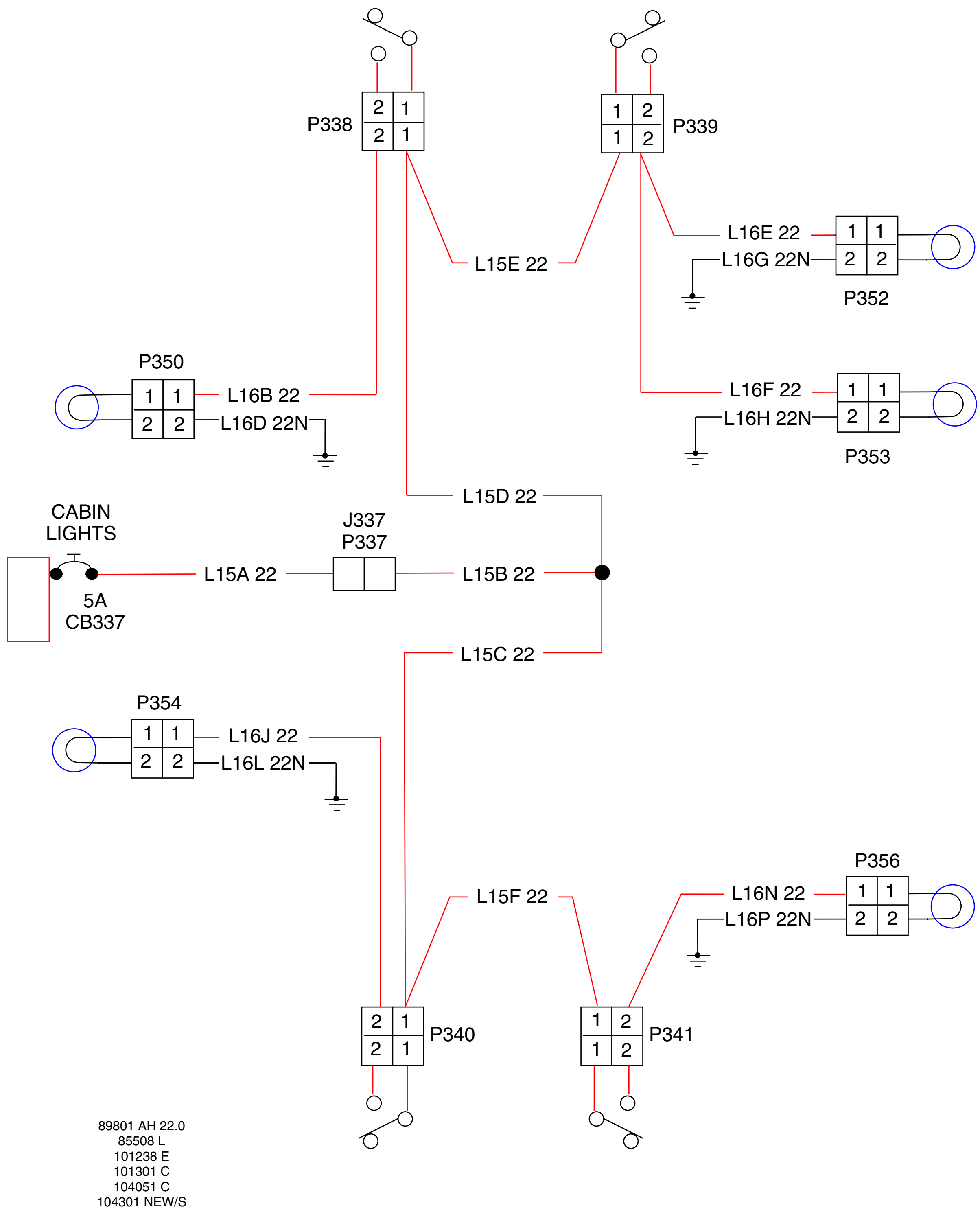
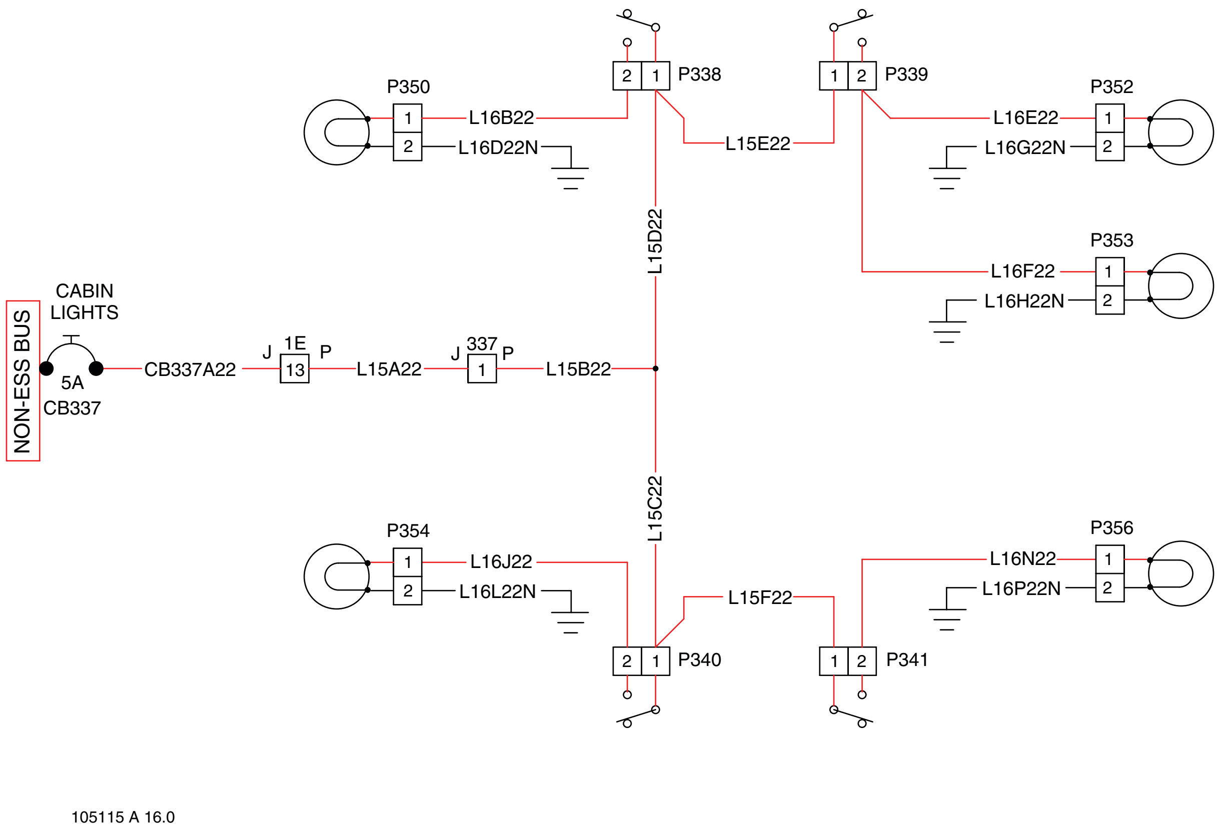
Item #	Designation	Description
1	CB337, CB55	Circuit Breaker - Cabin Lights (5 Amp)



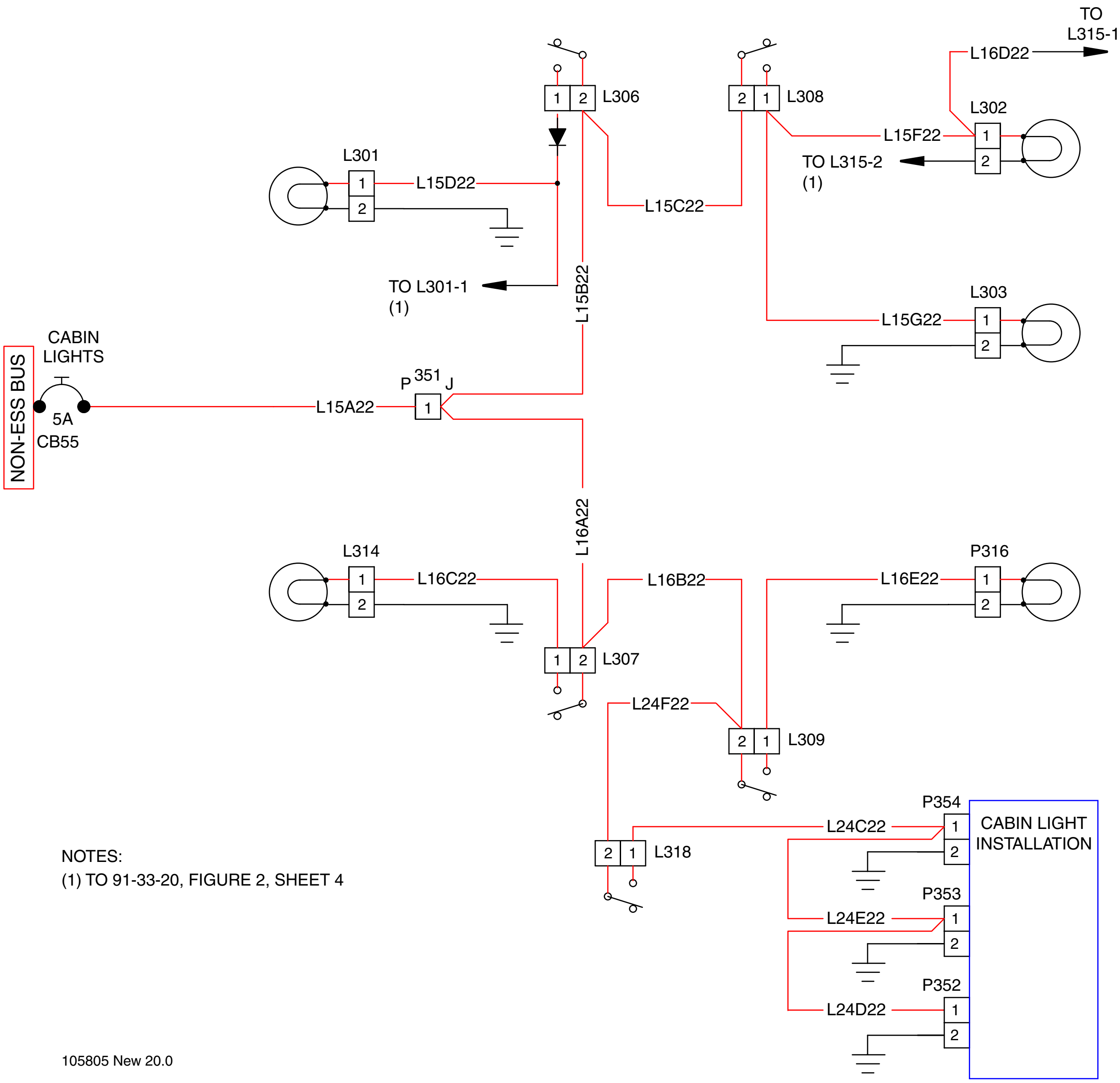
Cabin Lights
Figure 1 (Sheet 1 of 5)



Cabin Lights
Figure 1 (Sheet 2 of 5)

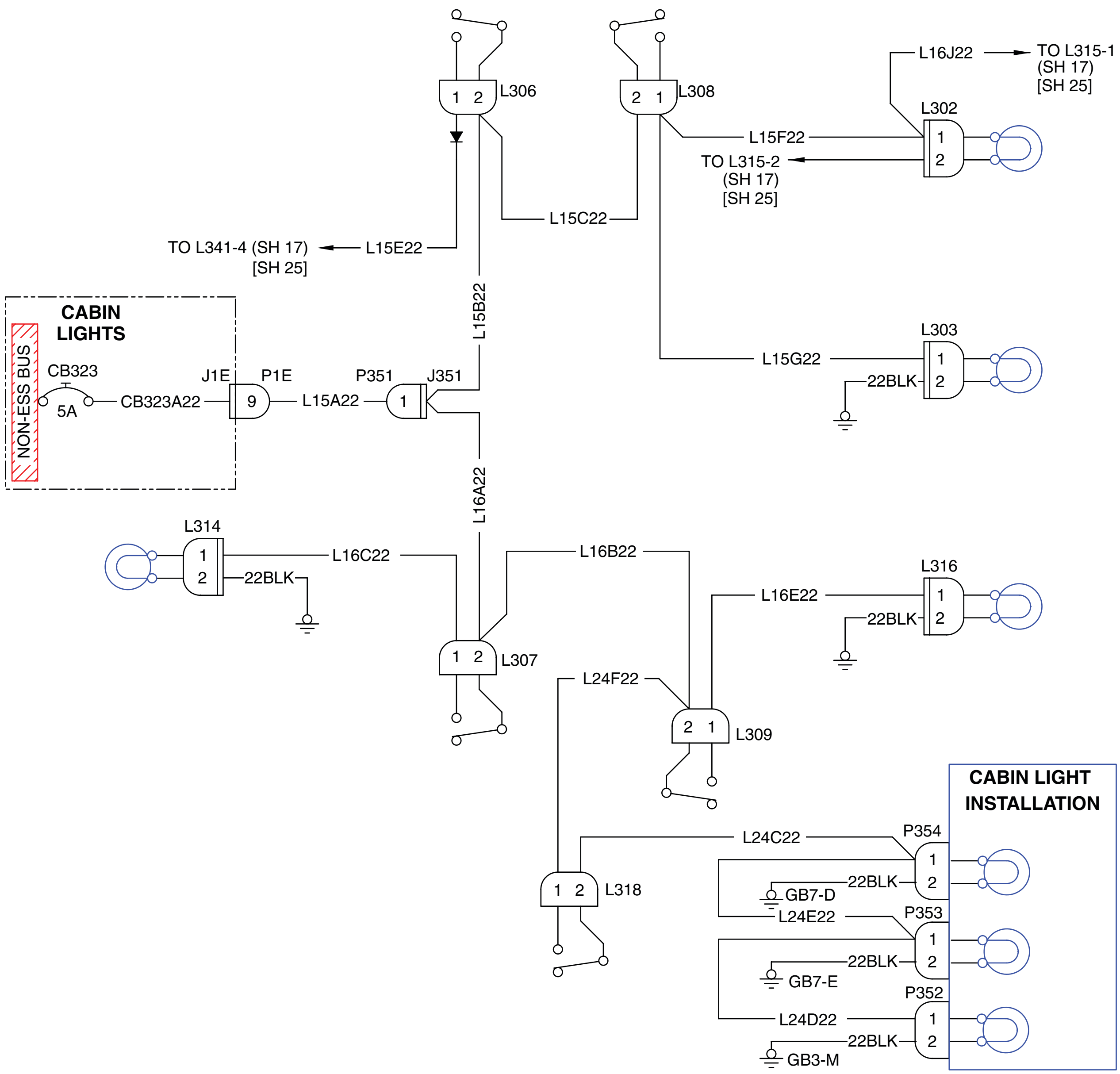


Cabin Lights
Figure 1 (Sheet 3 of 5)



Cabin Lights
Figure 1 (Sheet 4 of 5)

(105552 B 18)
[107975 NEW 26]



Cabin Lights
Figure 1 (Sheet 5 of 5)

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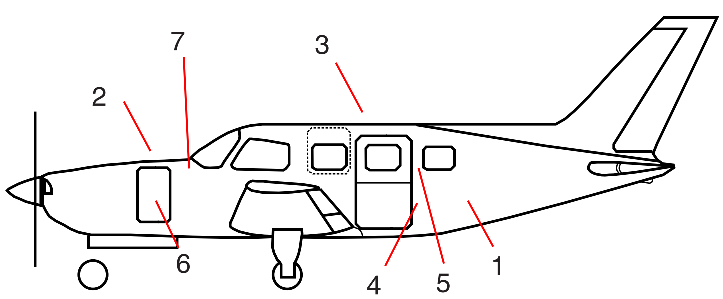
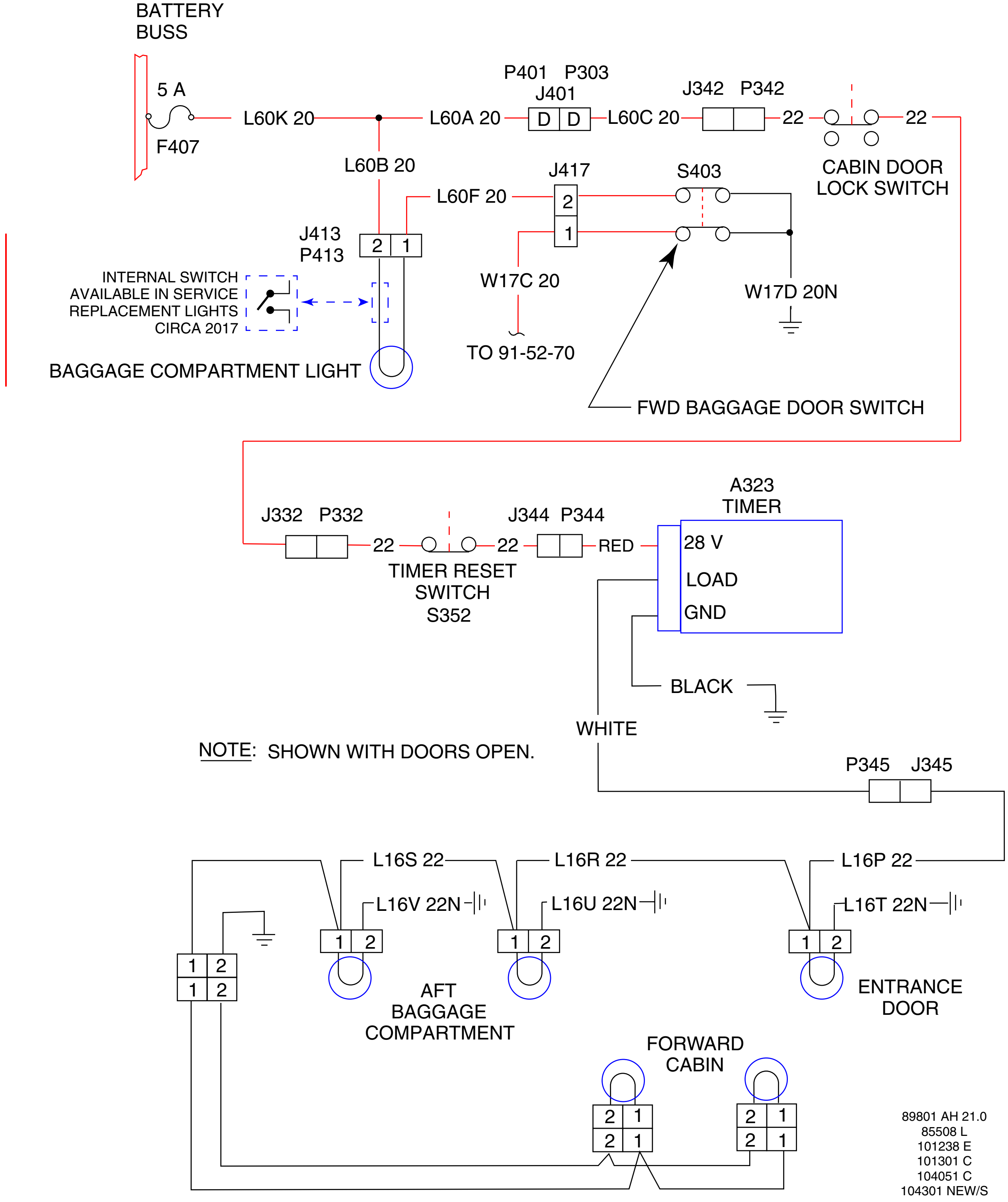
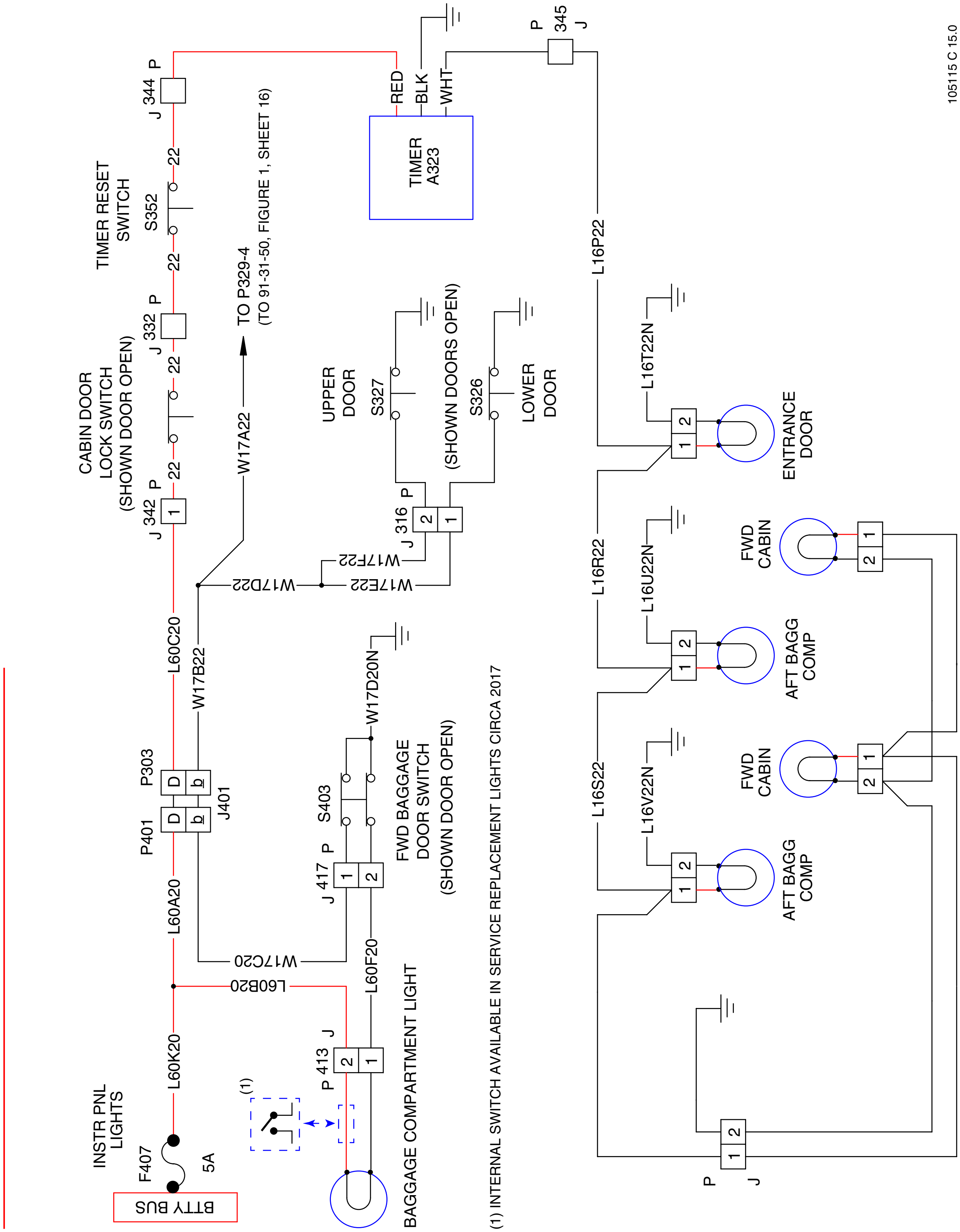
Item #	Designation	Description
1	A323	Timer
2		Baggage Compartment Light Assembly
3		Courtesy Light Assembly
4	S352	Timer Reset Switch
5		Cabin Door Lock Switch
6	S403	Baggage Door Switch
7	F407	Fuse (5A)



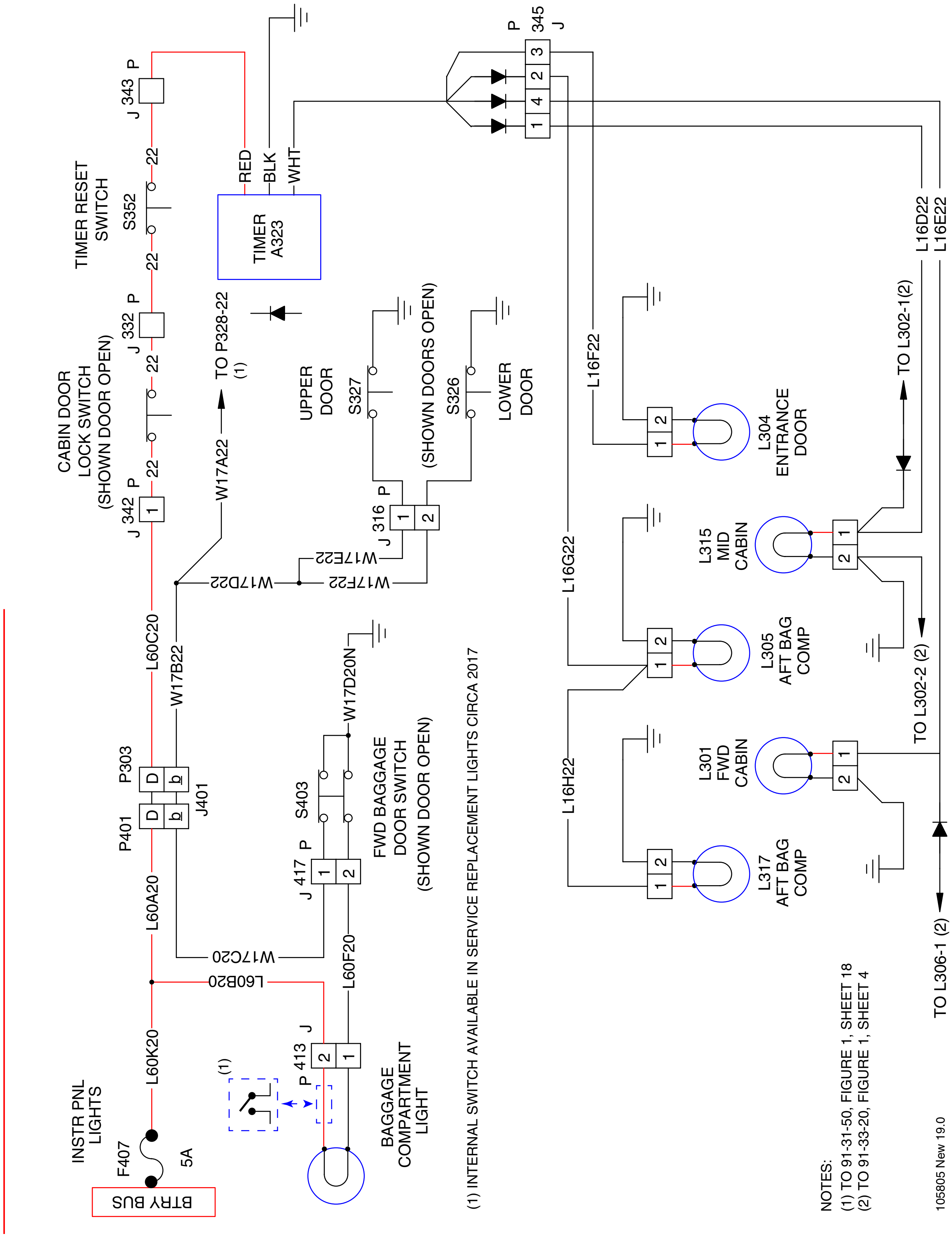
Courtesy Lights (Baggage Compartments, Door & Cabin)
Figure 2 (Sheet 1 of 6)



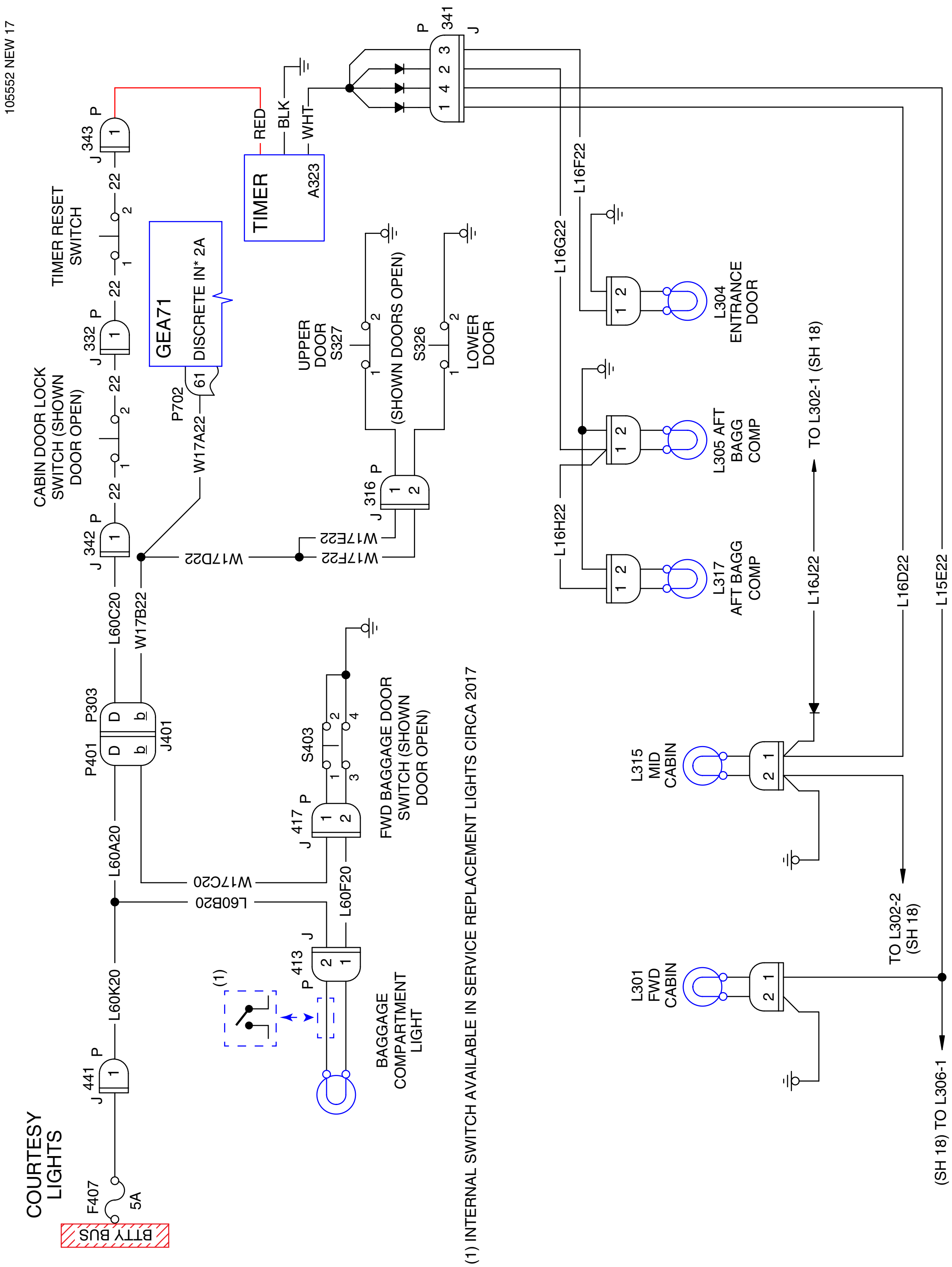
Courtesy Lights (Baggage Compartments, Door & Cabin)
Figure 2 (Sheet 2 of 6)



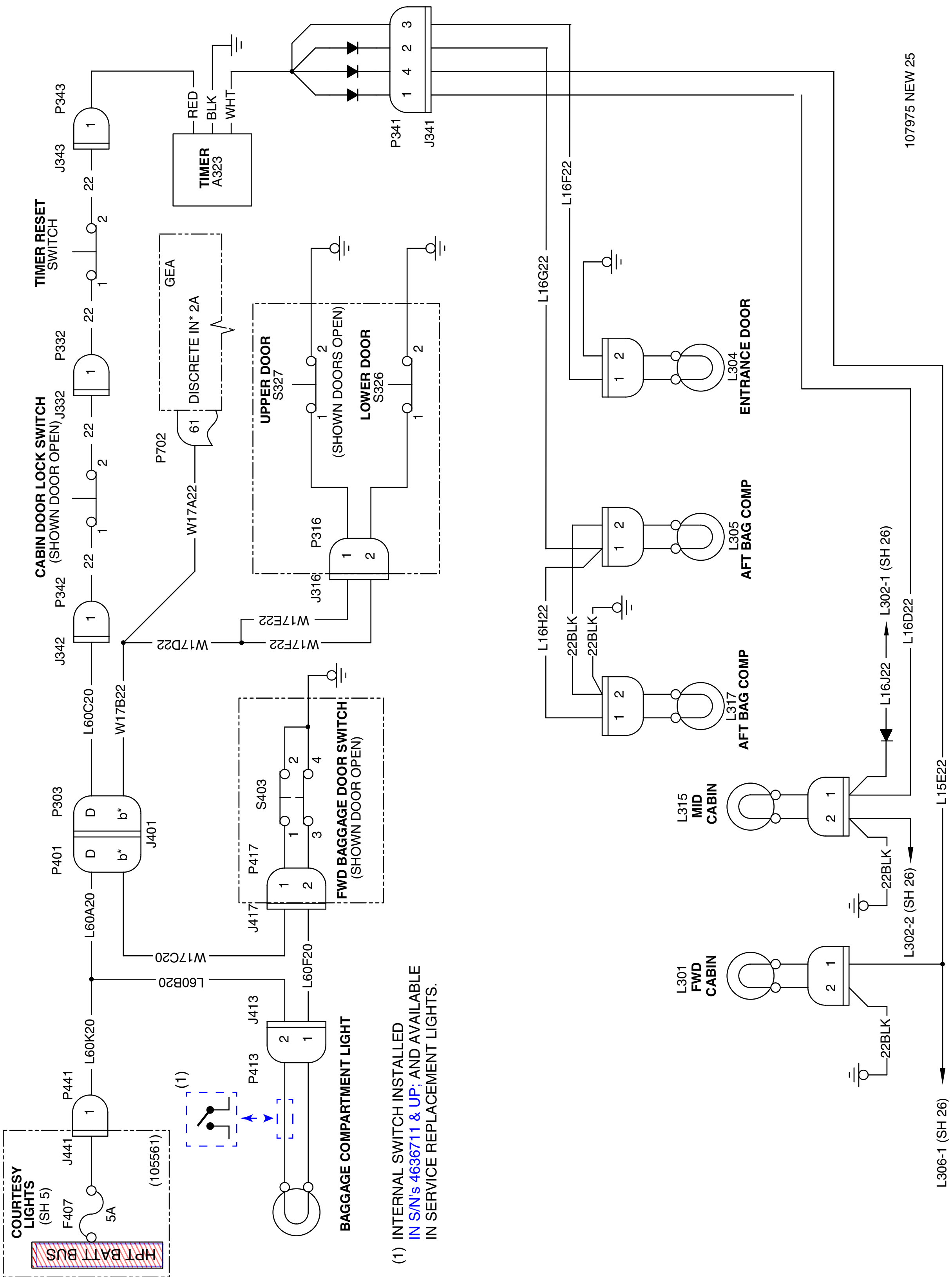
Courtesy Lights (Baggage Compartments, Door, & Cabin)
Figure 2 (Sheet 3 of 6)



Courtesy Lights (Baggage Compartments, Door, & Cabin)
Figure 2 (Sheet 4 of 6)

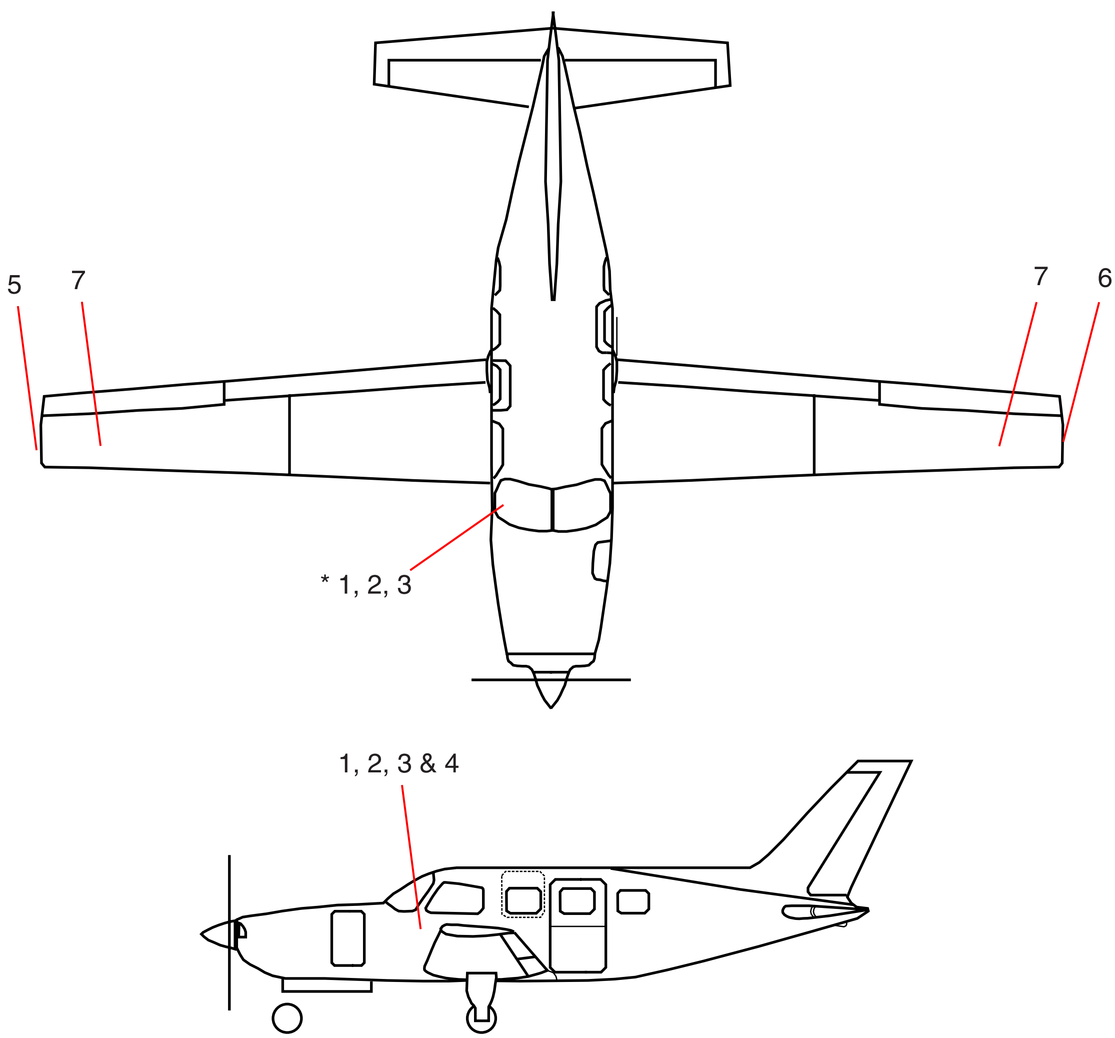


Courtesy Lights (Baggage Compartments, Door, & Cabin)
Figure 2 (Sheet 5 of 6)

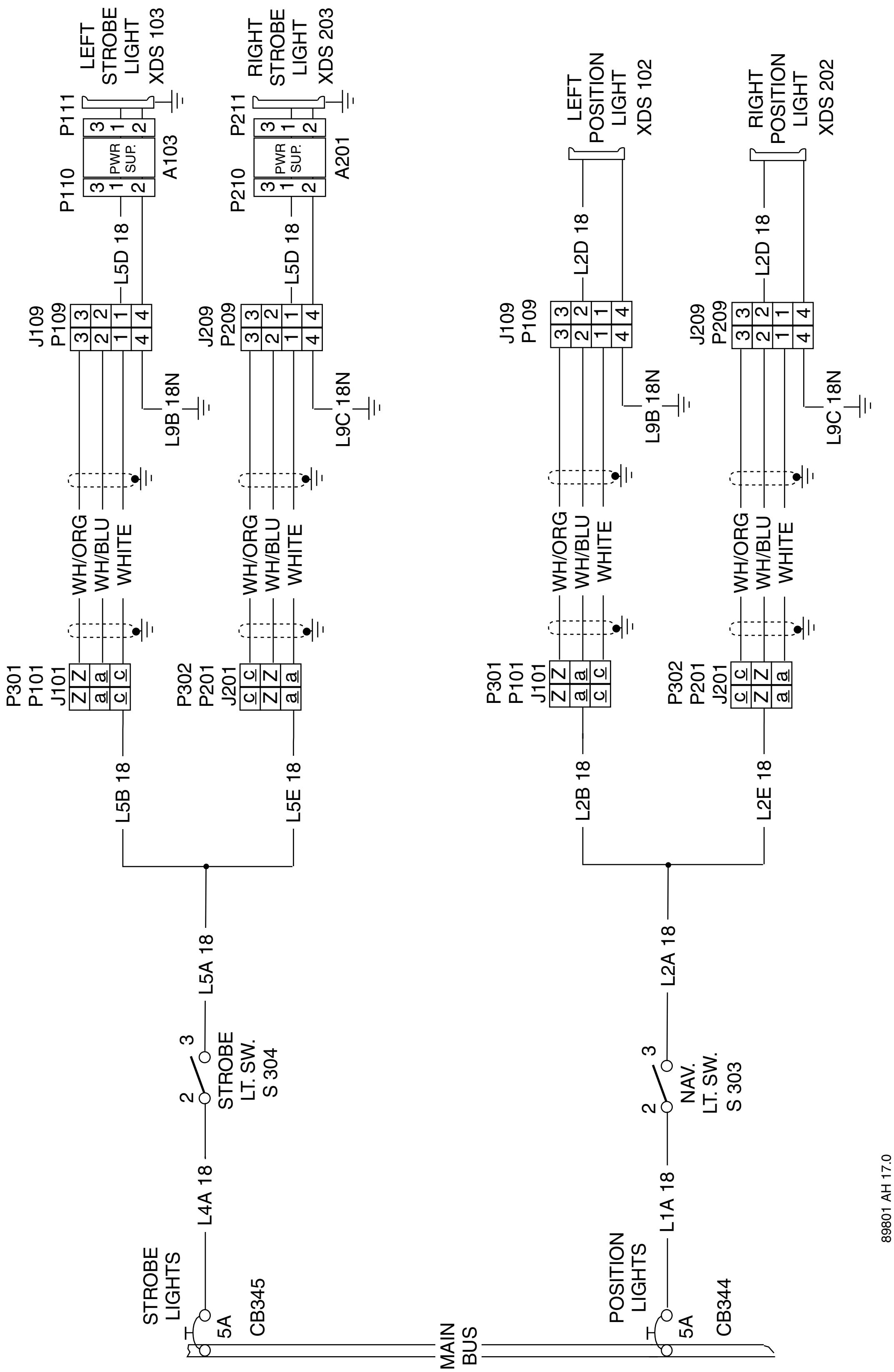


Courtesy Lights (Baggage Compartments, Door, & Cabin)
Figure 2 (Sheet 6 of 6)

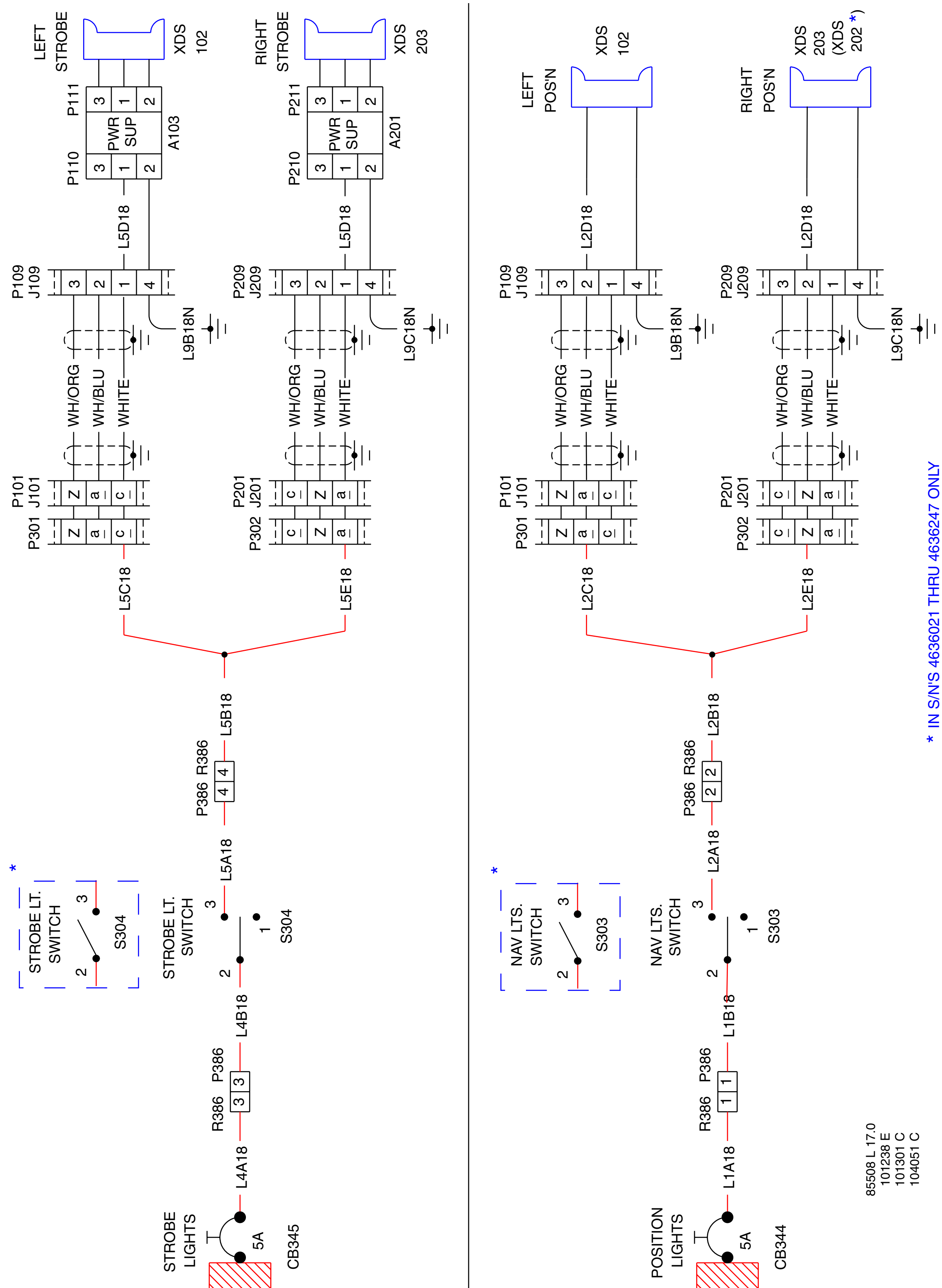
Item #	Designation	Description
1	CB344, CB15*	Circuit Breaker - Position Lights (5 Amp)
2	CB345, CB16*	Circuit Breaker - Strobe Lights (5 or 7.5 Amp)
3	CB347, CB18*	Circuit Breaker - Taxi Lights (5 Amp)
4	S303, S304	Nav/Pos & Strobe Lights Switch
5	XDS 202, XDS 203	Nav/Pos & Strobe Light, Right
6	XDS 102, XDS 103	Nav/Pos & Strobe Light, Left
7	A103, A201	Strobe Light Power Supply



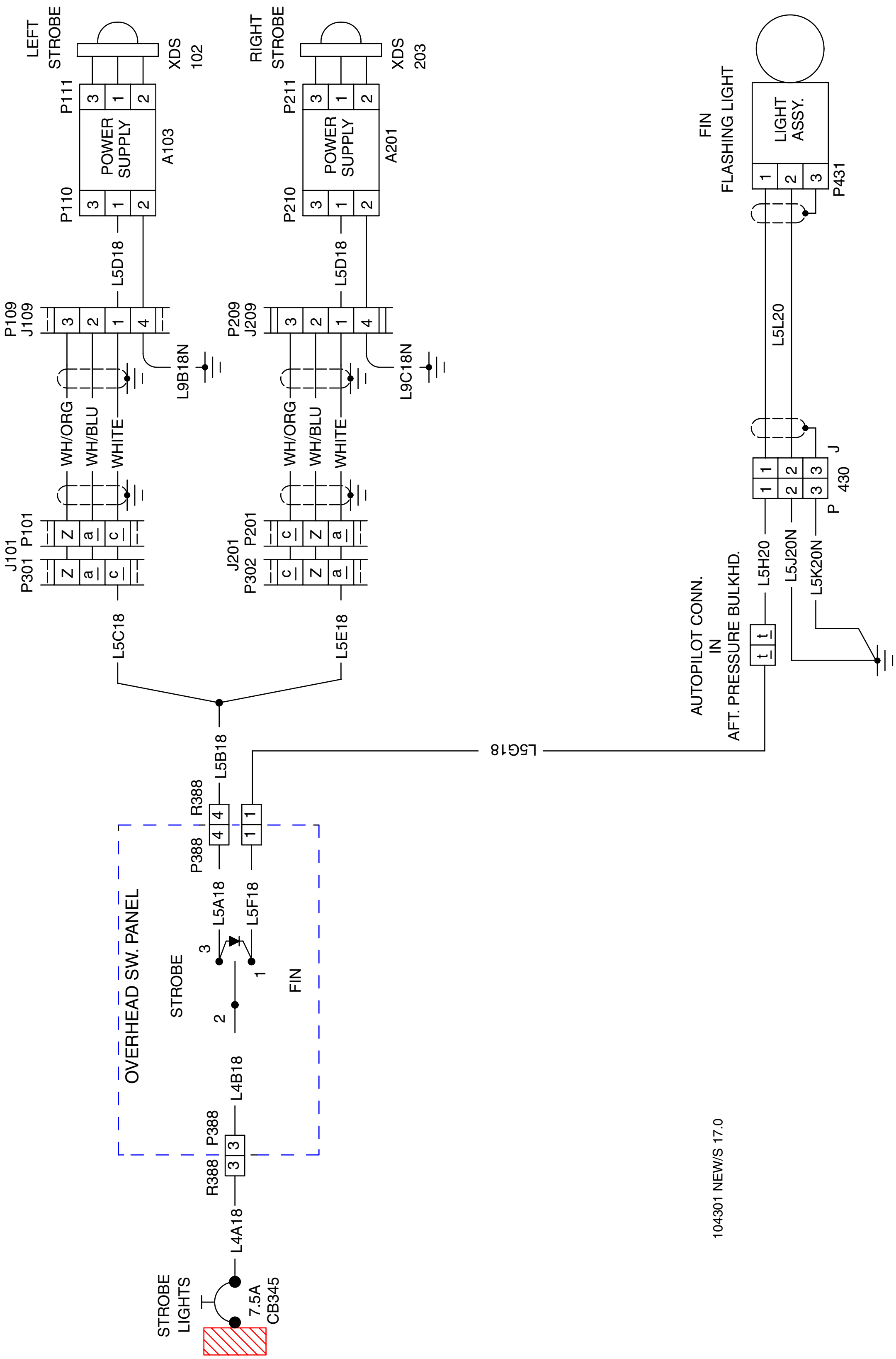
Position and Strobe Lights
Figure 1 (Sheet 1 of 9)



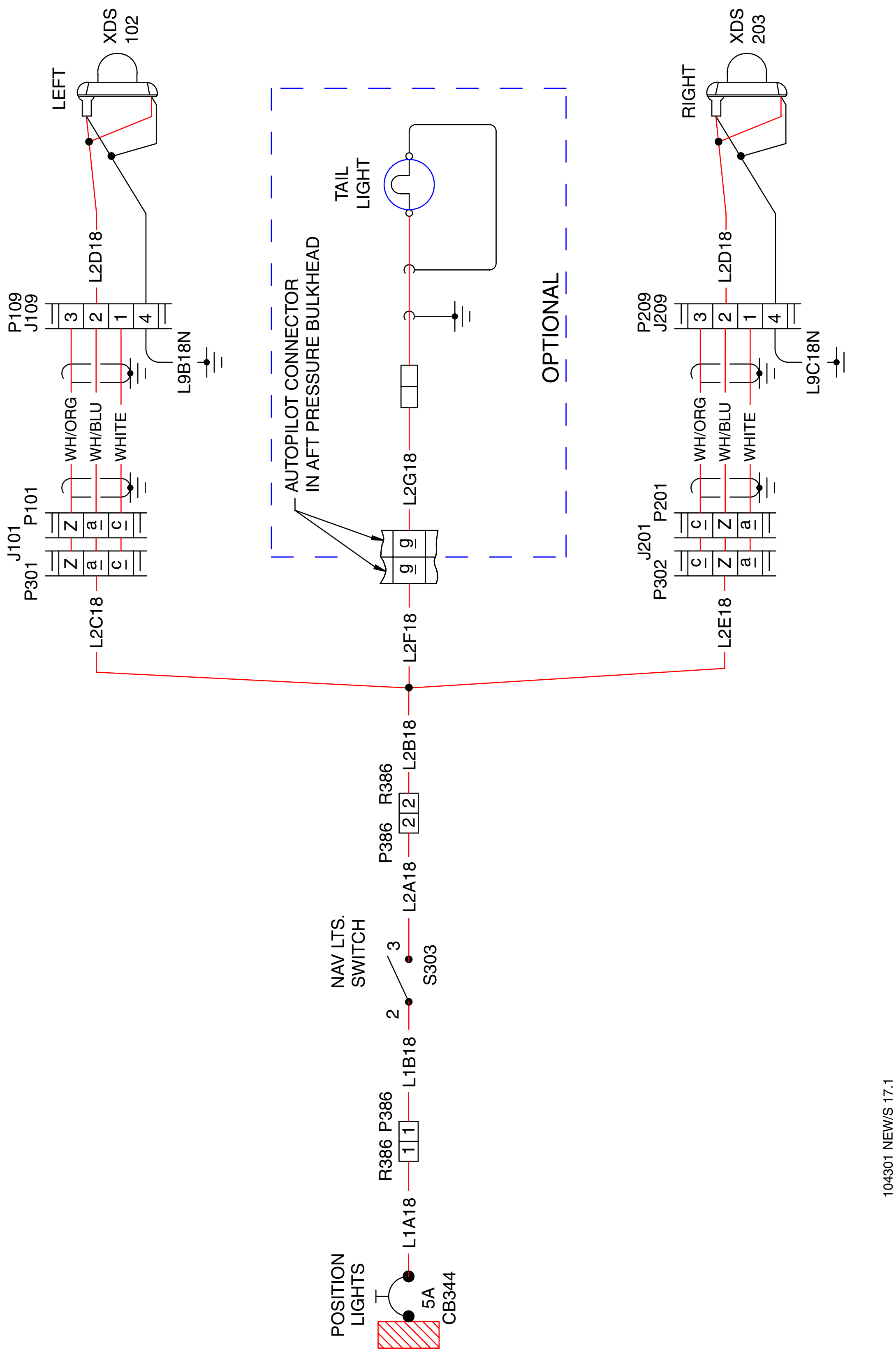
Position and Strobe Lights
Figure 1 (Sheet 2 of 9)



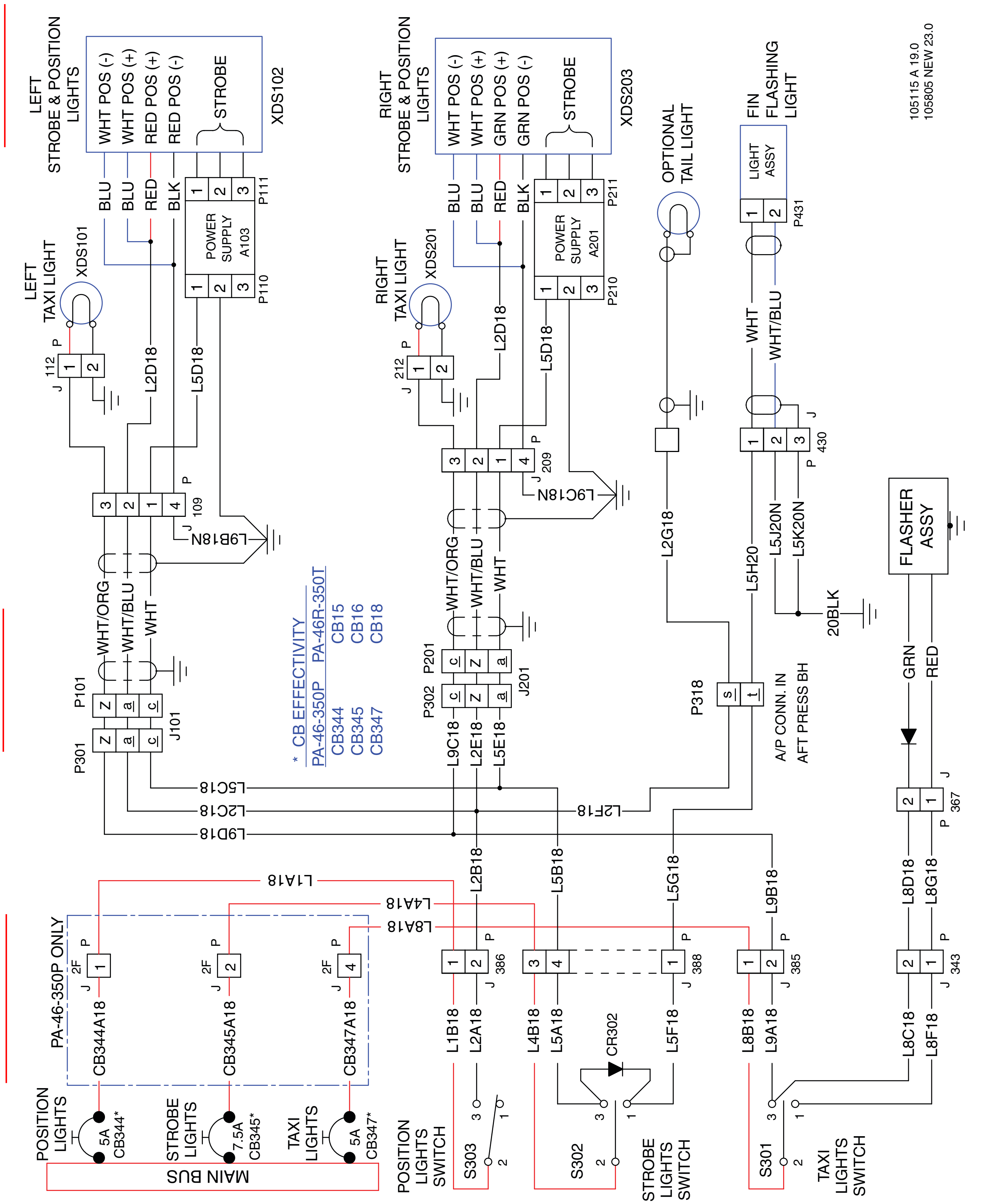
Position and Strobe Lights
Figure 1 (Sheet 3 of 9)



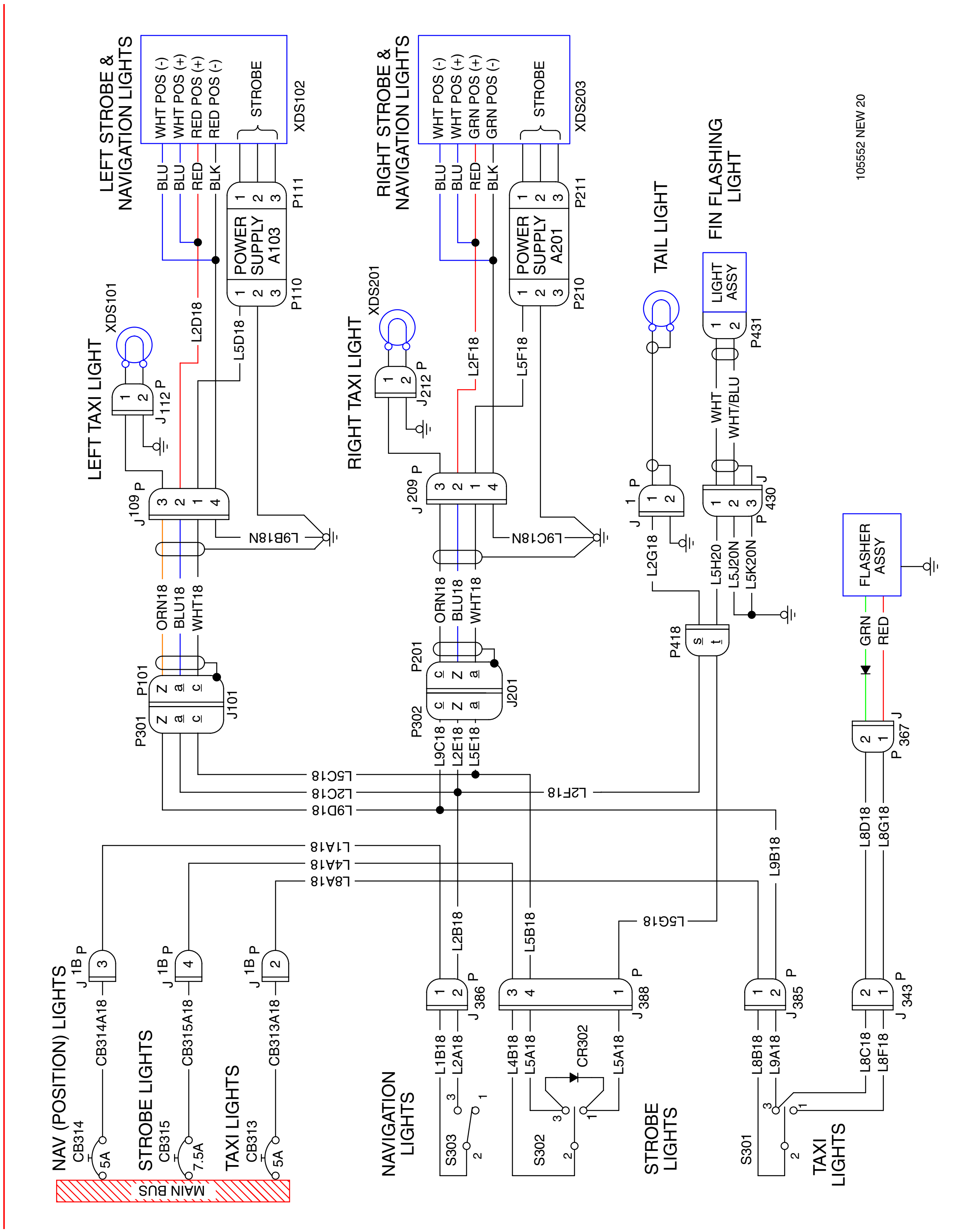
Position and Strobe Lights
Figure 1 (Sheet 4 of 9)



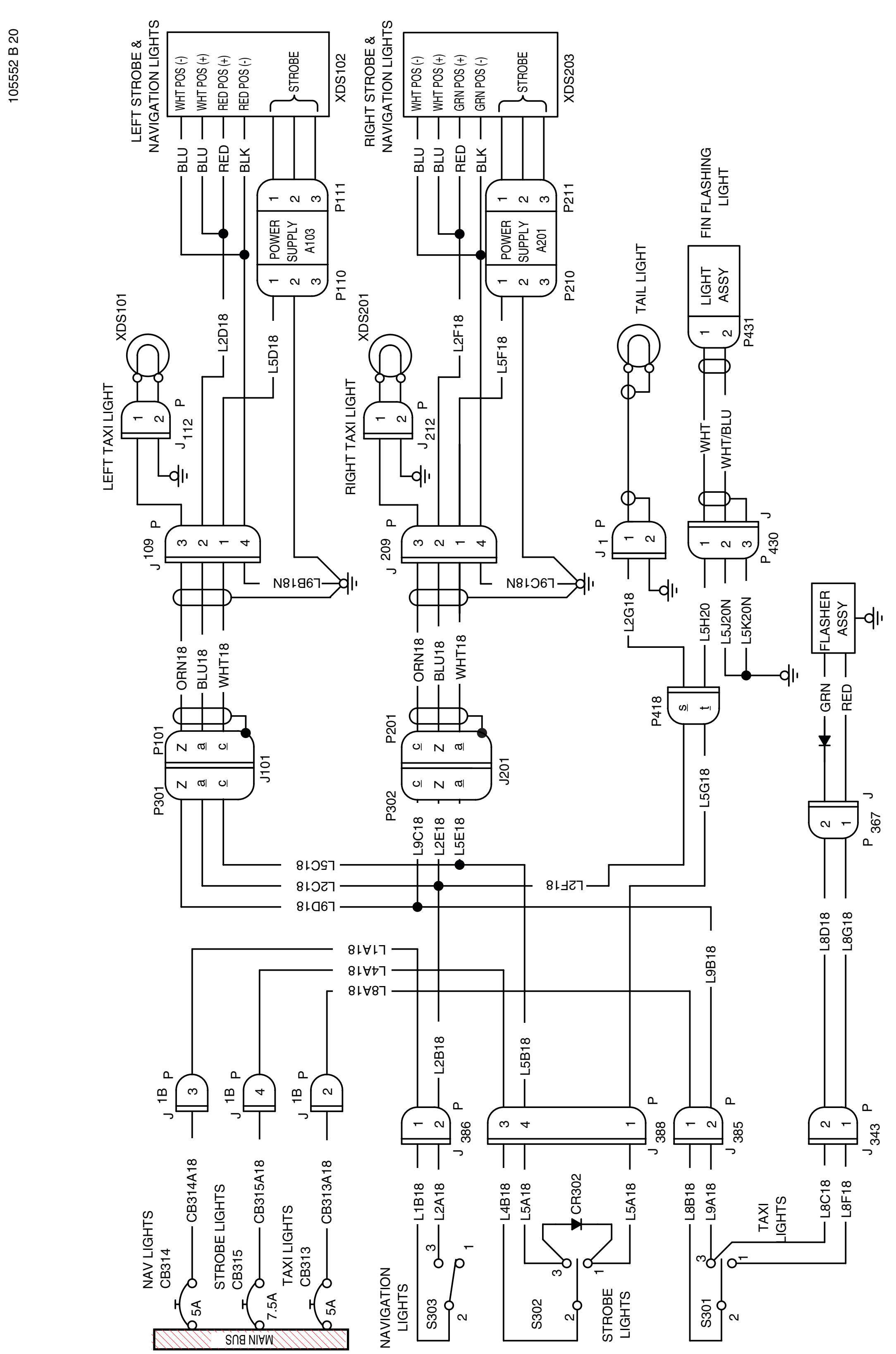
Position and Strobe Lights
Figure 1 (Sheet 5 of 9)



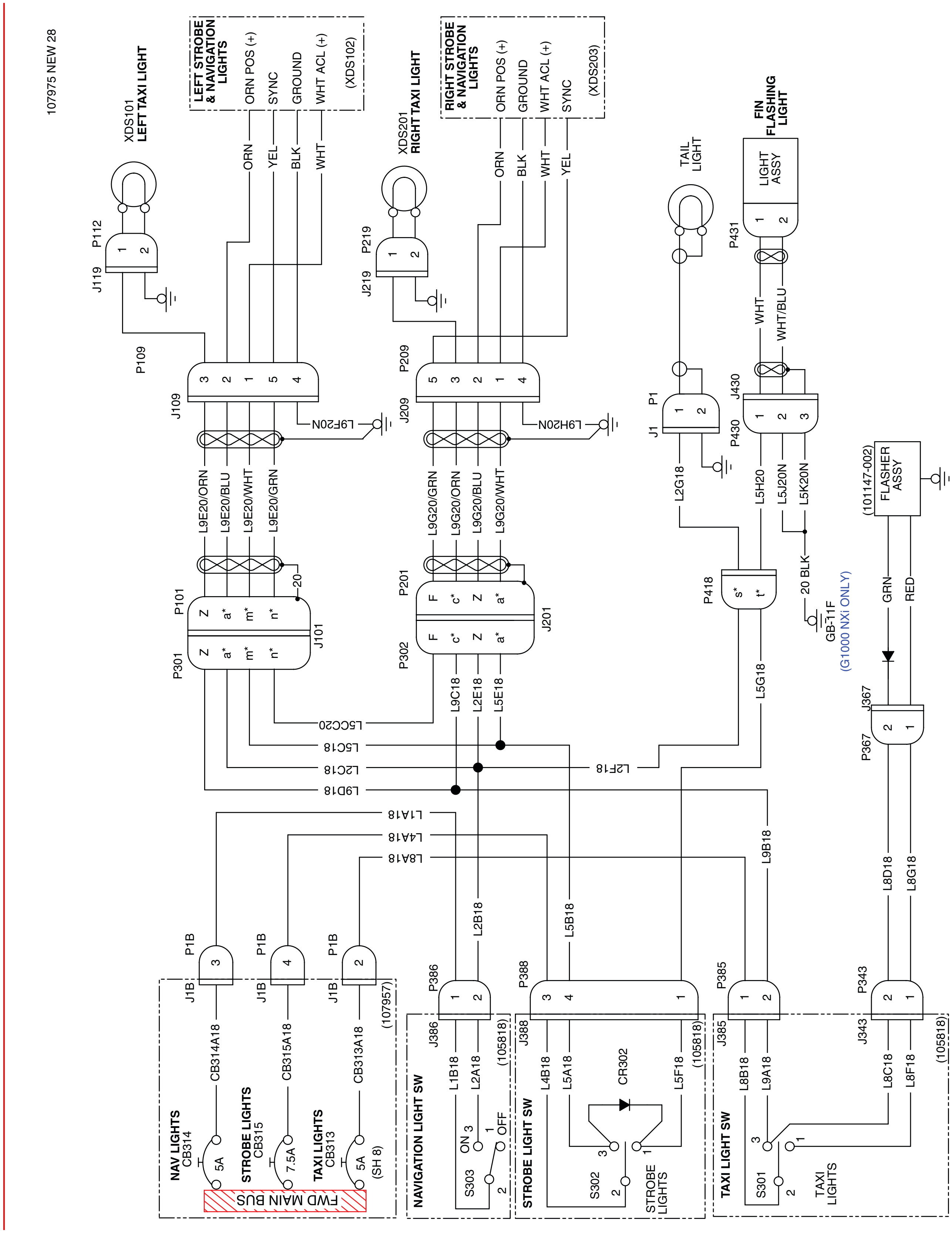
Position and Strobe Lights
Figure 1 (Sheet 6 of 9)



Position and Strobe Lights
Figure 1 (Sheet 7 of 9)



Position and Strobe Lights
Figure 1 (Sheet 8 of 9)



Position and Strobe Lights
Figure 1 (Sheet 9 of 9)

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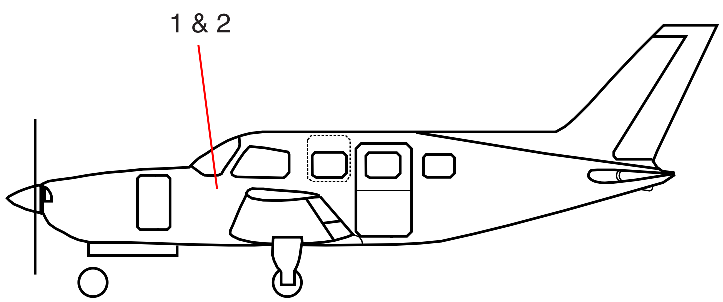
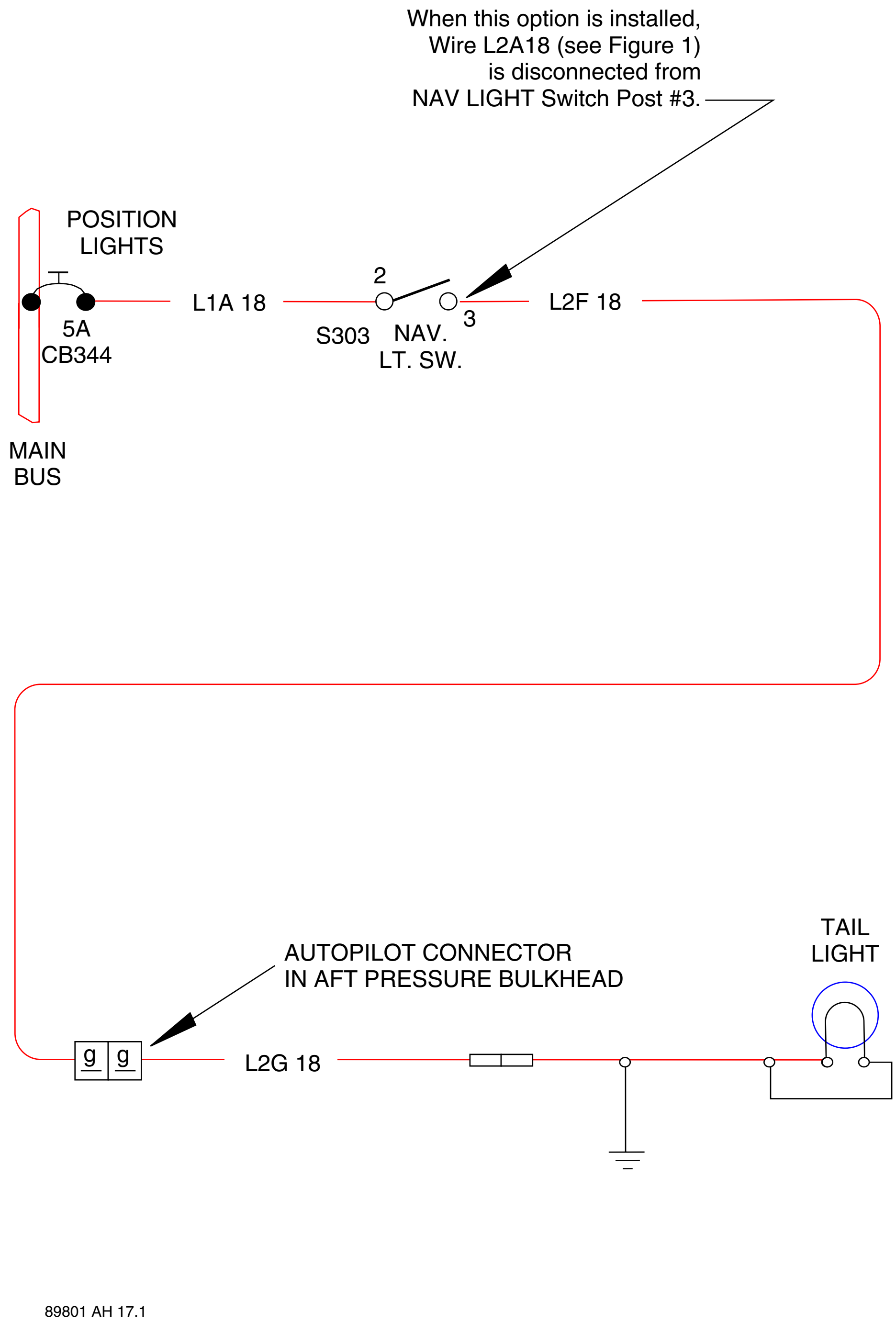
Item #	Designation	Description
1	CB344	Circuit Breaker - Position Lights (5 Amp)
2	S303	Nav Lights Switch

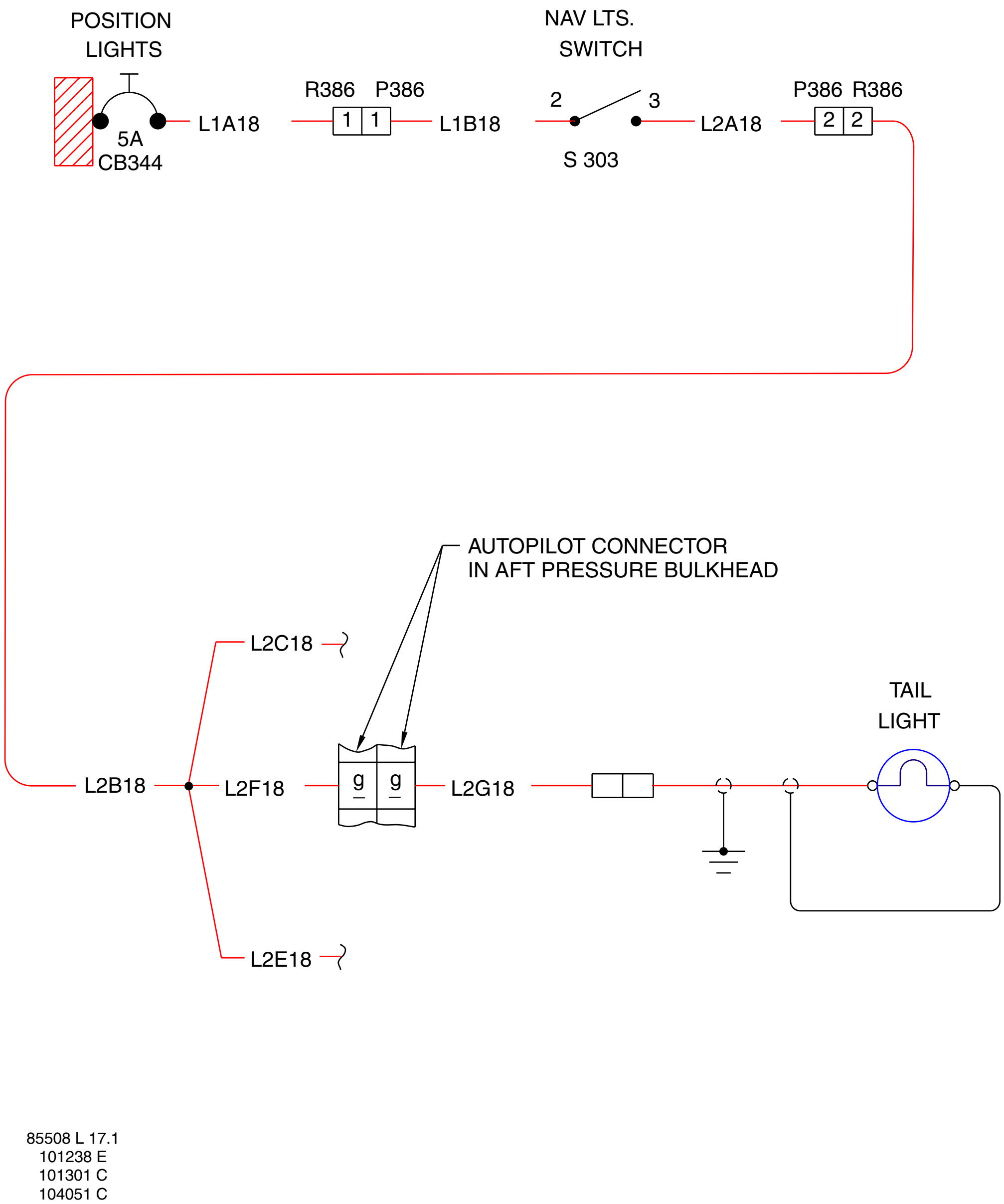
In S/N's 4636299 and 4636314 & up and
S/N's 4692001 & up, see Figure 1.



Tail Light (Optional)
Figure 2 (Sheet 1 of 3)



Tail Light (Optional)
Figure 2 (Sheet 2 of 3)



Tail Light (Optional)
Figure 2 (Sheet 3 of 3)

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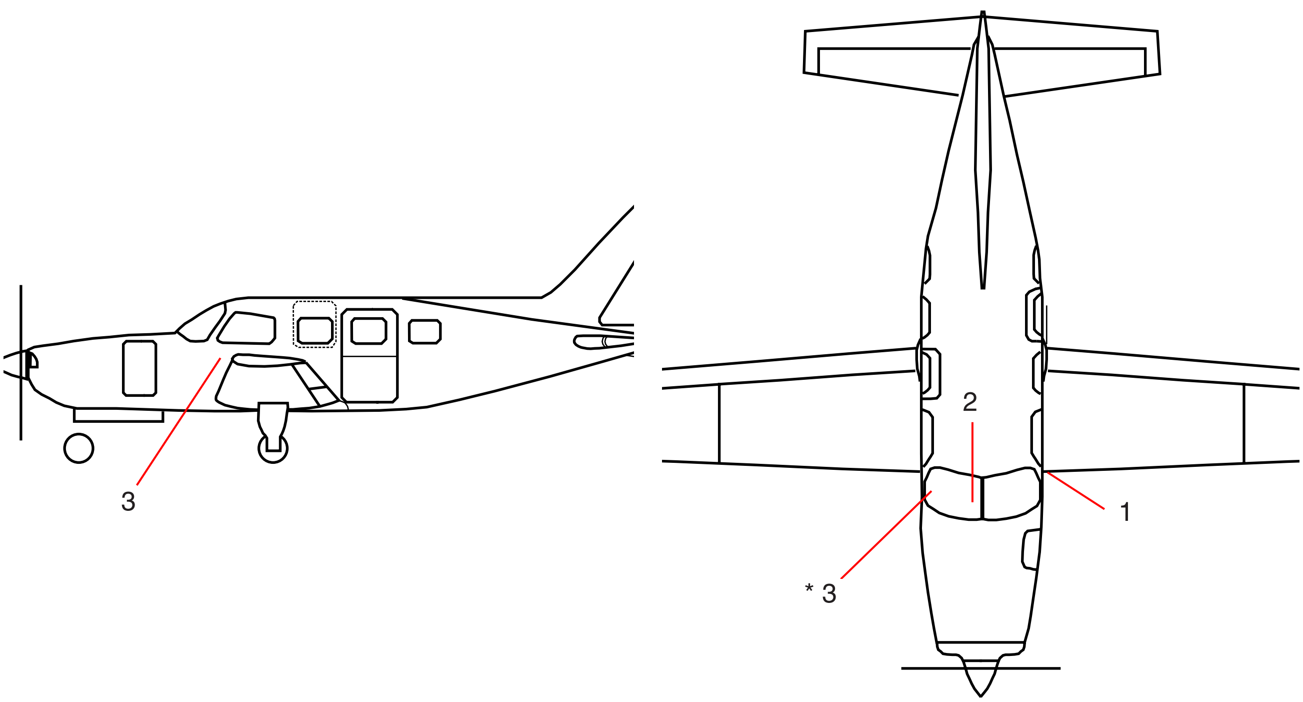
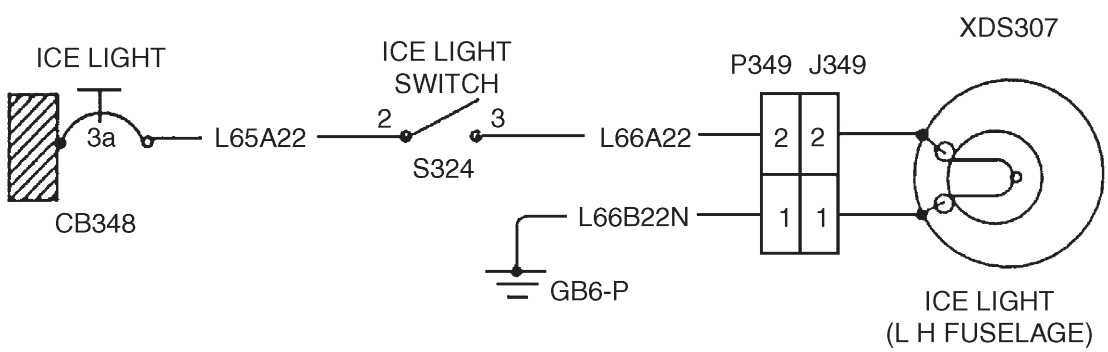
Item #	Designation	Description
1	XDS307	Ice Light
2	S324	Ice Light Switch
3	CB348, CB19*	Circuit Breaker - Ice Light (3 Amp)

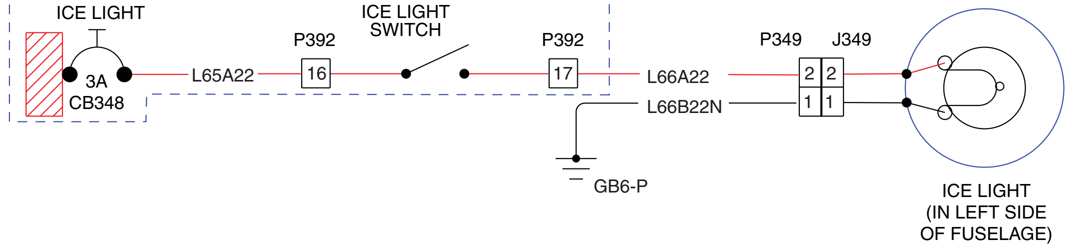


Wing Inspection (Ice) Light (Optional)
Figure 3 (Sheet 1 of 4)

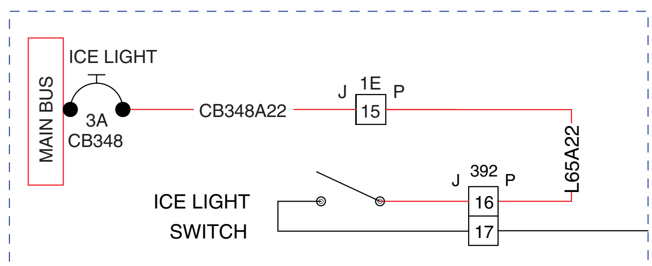


89801 AH 24.0
85508 L

Wing Inspection (Ice) Light (Optional)
Figure 3 (Sheet 2 of 4)

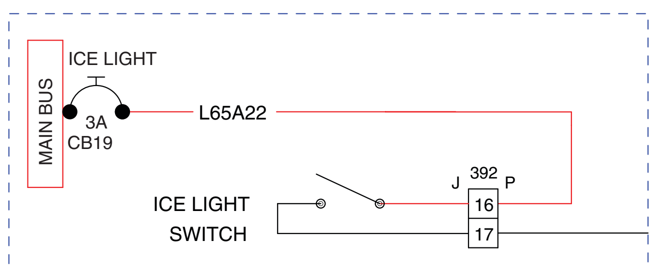


101238 E 24.0
101301 C
104051 C
104301 NEW/S



S/N'S 4636375–4636459, 4636461–4636462, 4636481

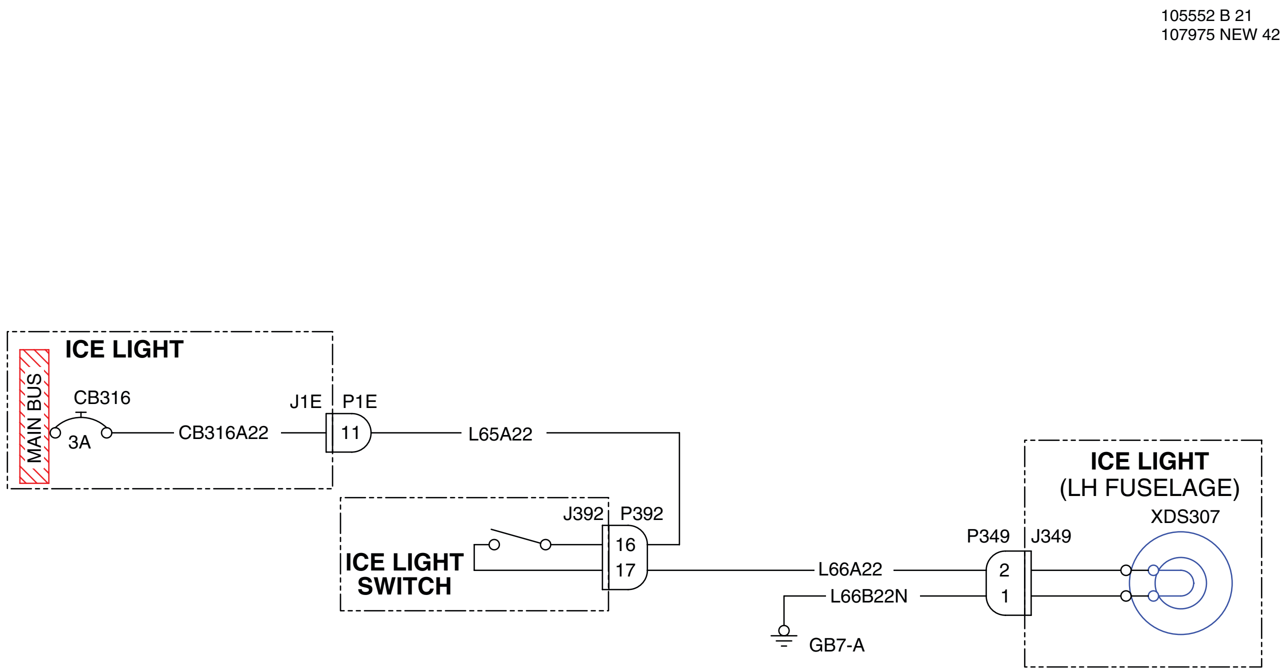
105115 A 18.0



S/N'S 4692001–4692133, 4692141, 4692149, 4692153

105805 New 22.0

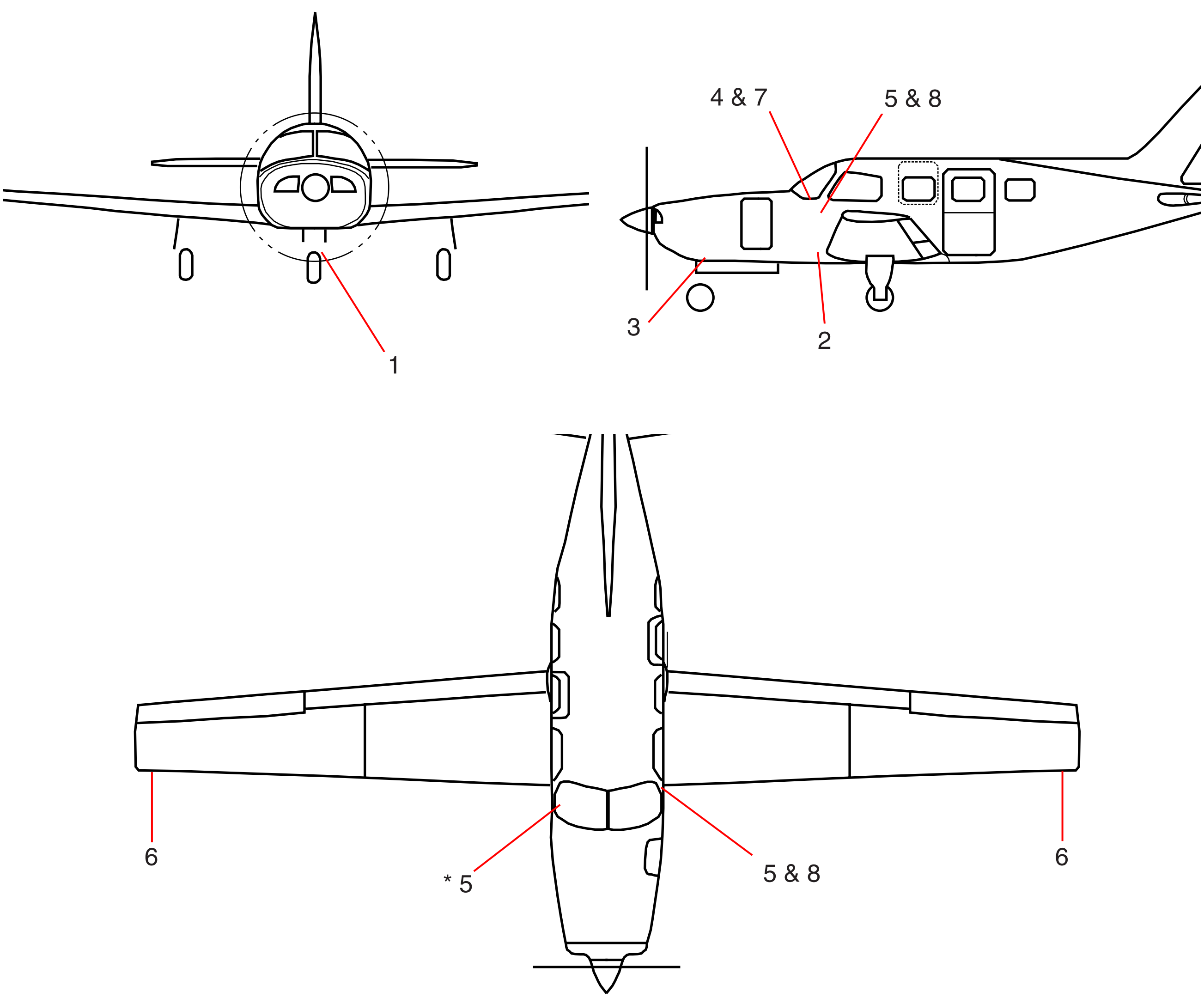
Wing Inspection (Ice) Light (Optional)
Figure 3 (Sheet 3 of 4)



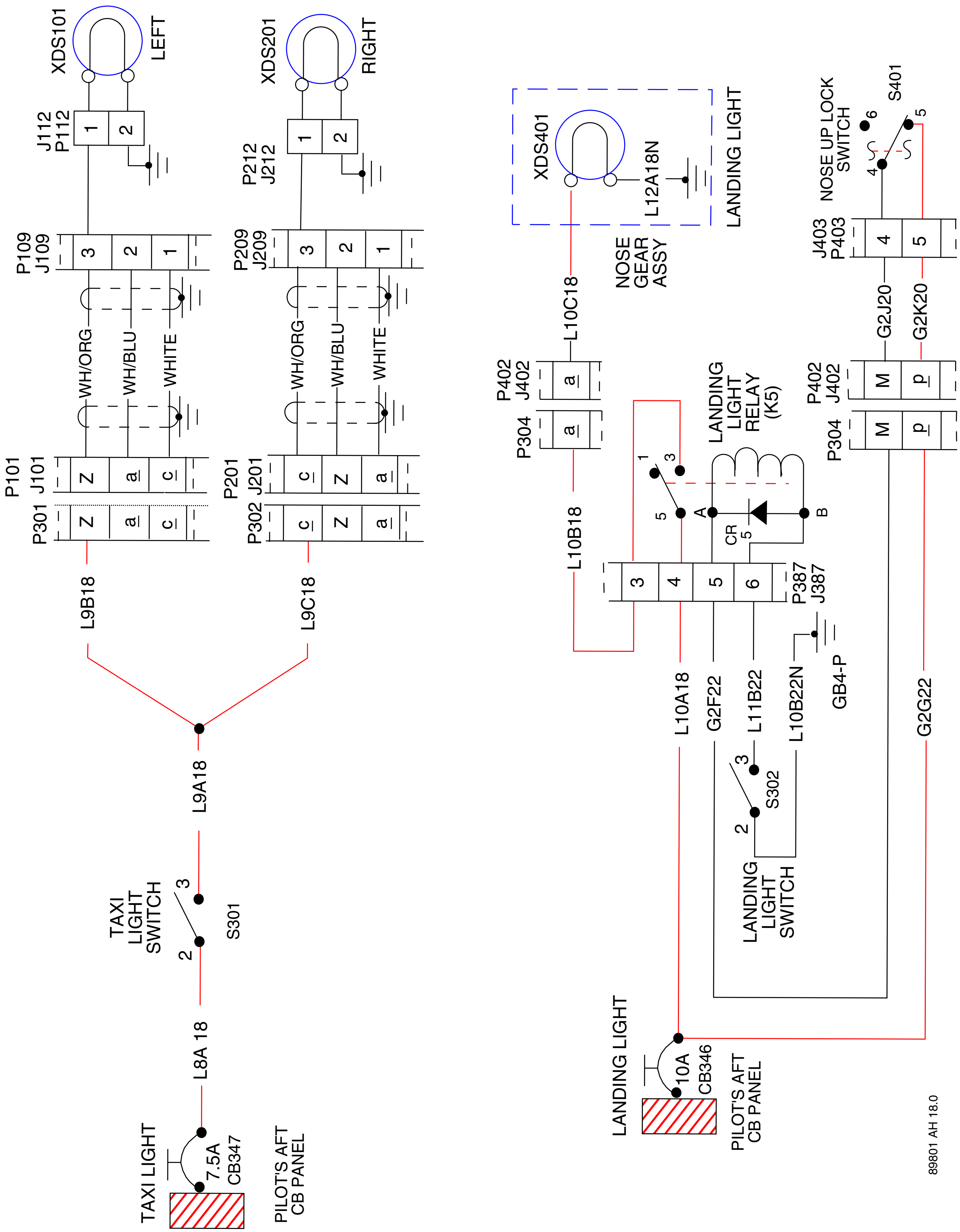
Wing Inspection (Ice) Light (Optional)
Figure 3 (Sheet 4 of 4)

Item #	Designation	Description
1	XDS401	Landing Light
2	K5	Landing Light Relay
3	S401	Nose Gear Up Lock Switch
4	S302	Landing Light Switch
5	CB346, CB17*	Circuit Breaker - Landing Light (10 Amp.)
6	XDS101, XDS201	Taxi Lights
7	S301	Taxi Lights Switch
8	CB347	Circuit Breaker - Taxi Light (7.5 or 5 Amp.)

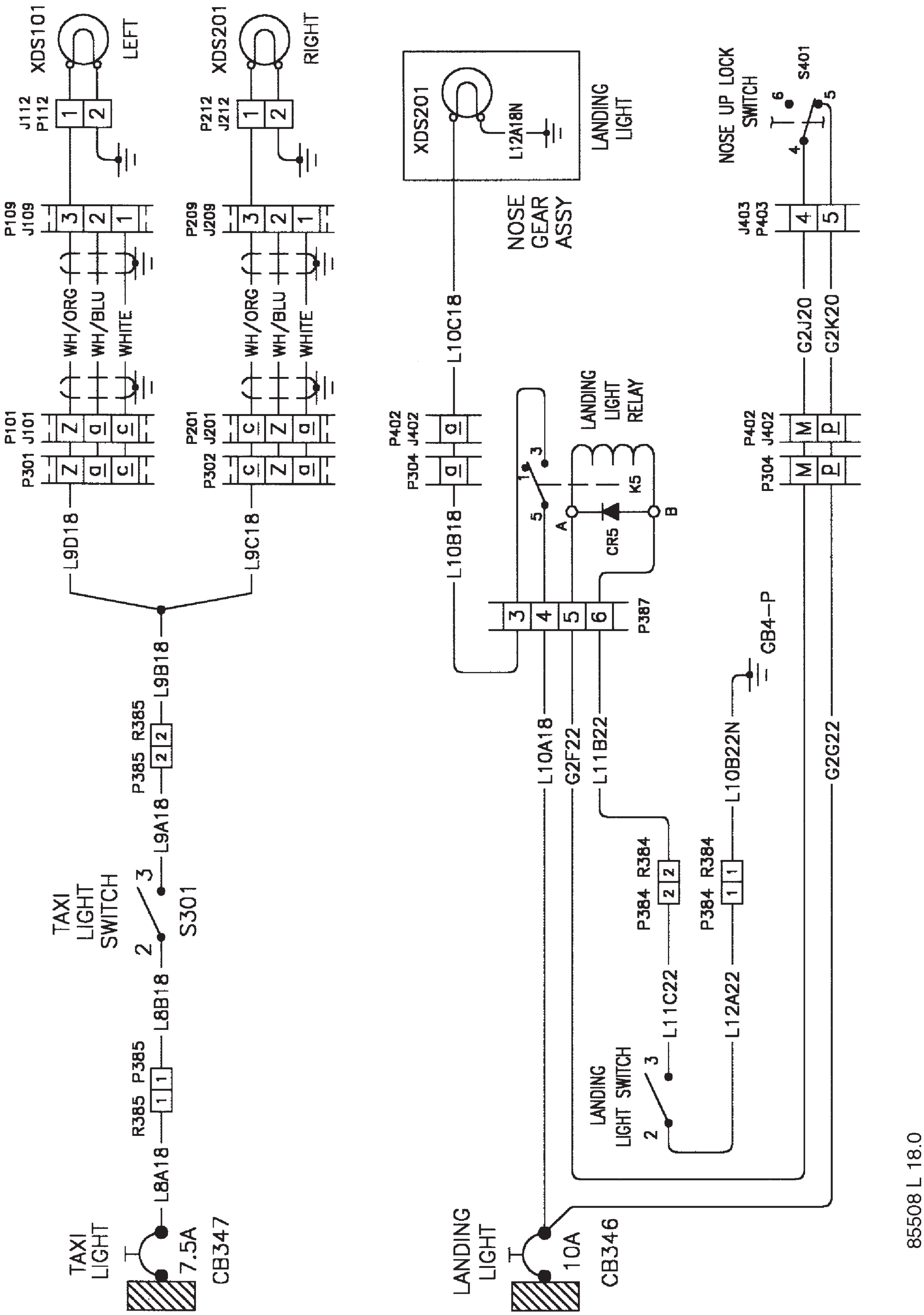
In S/N's 4636375 and up, and
S/N's 4692001 and up, see Figure 1, for Taxi Lights.



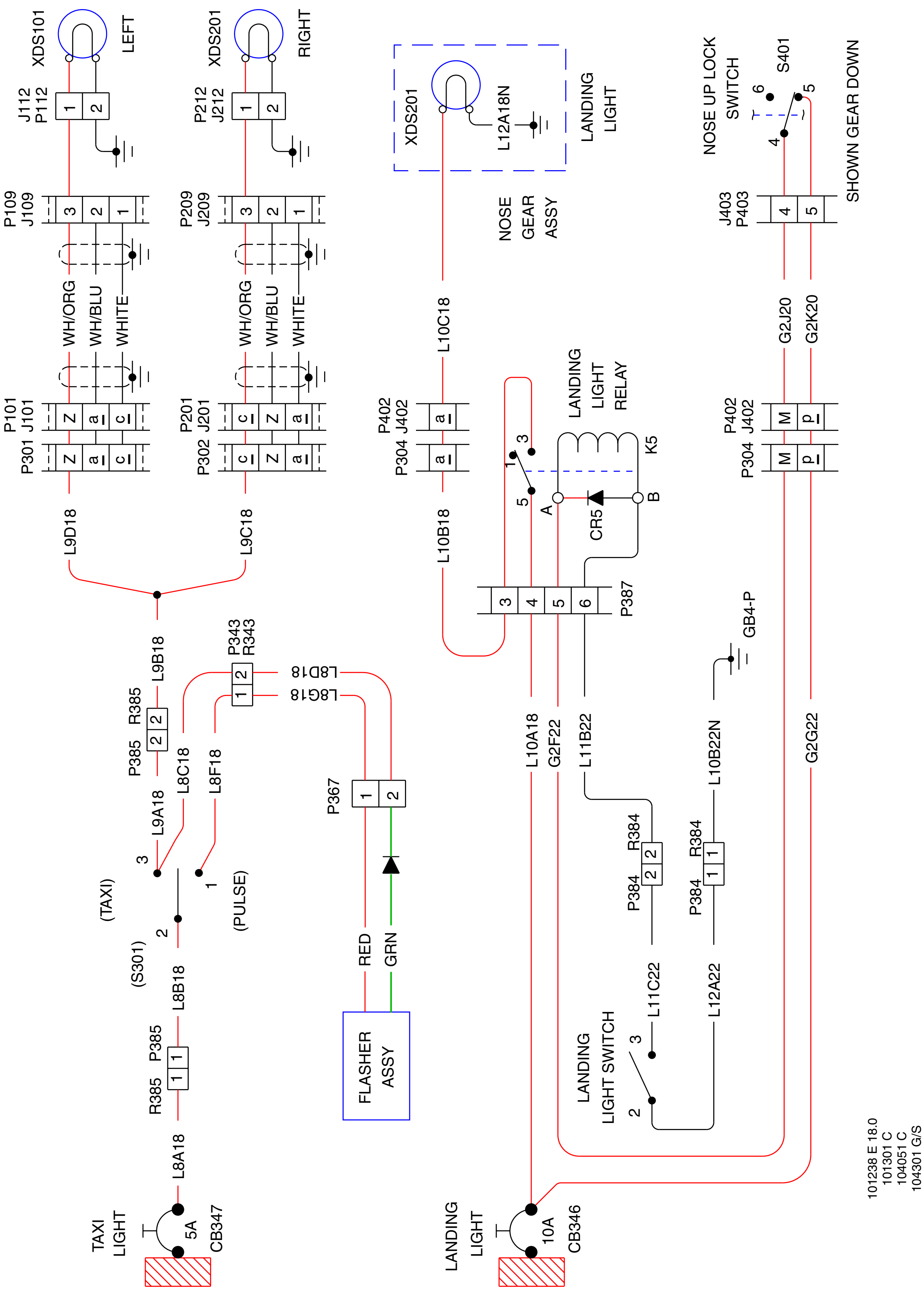
Landing and Taxi Lights
Figure 4 (Sheet 1 of 7)



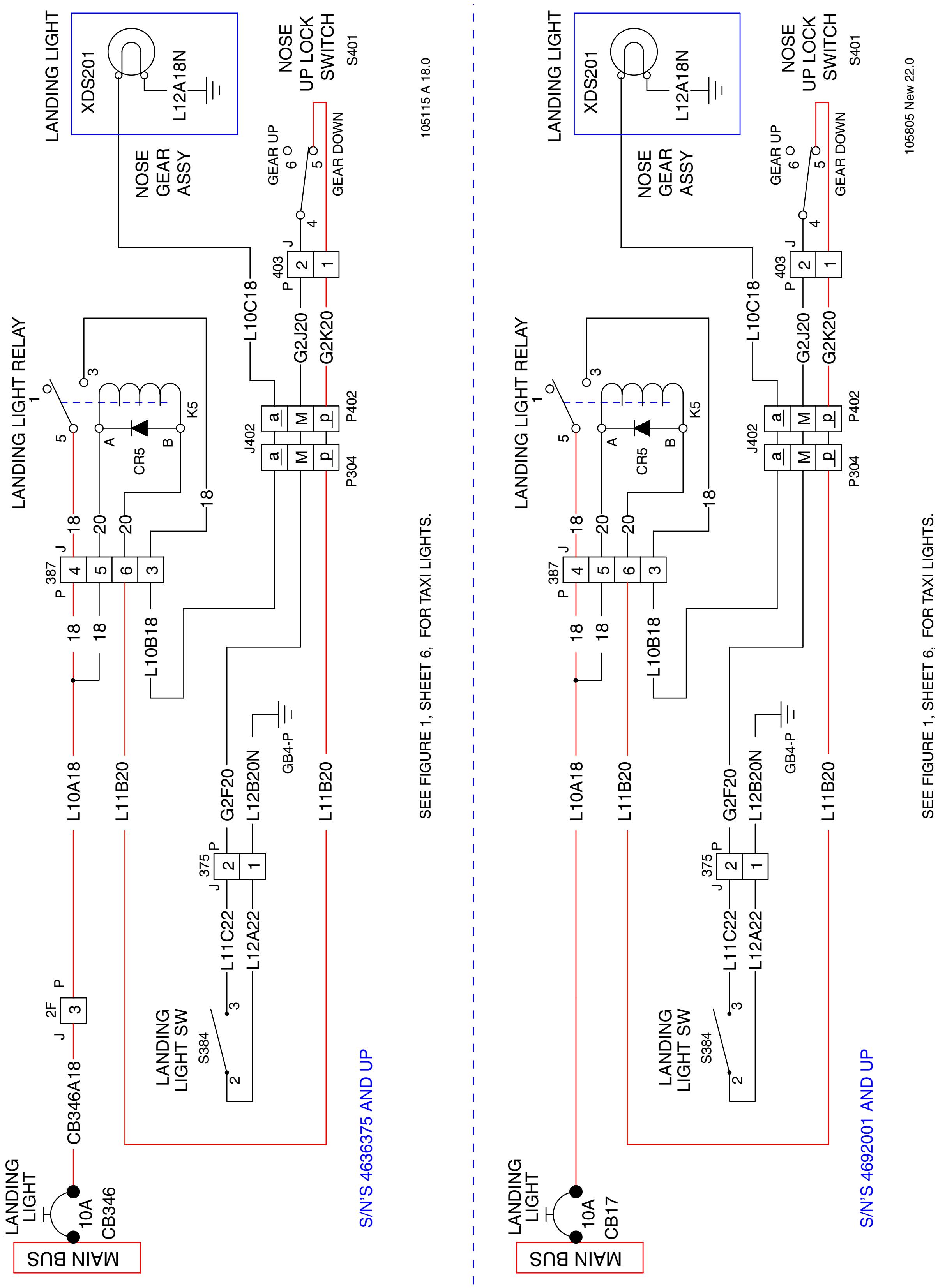
Landing and Taxi Lights
Figure 4 (Sheet 2 of 7)



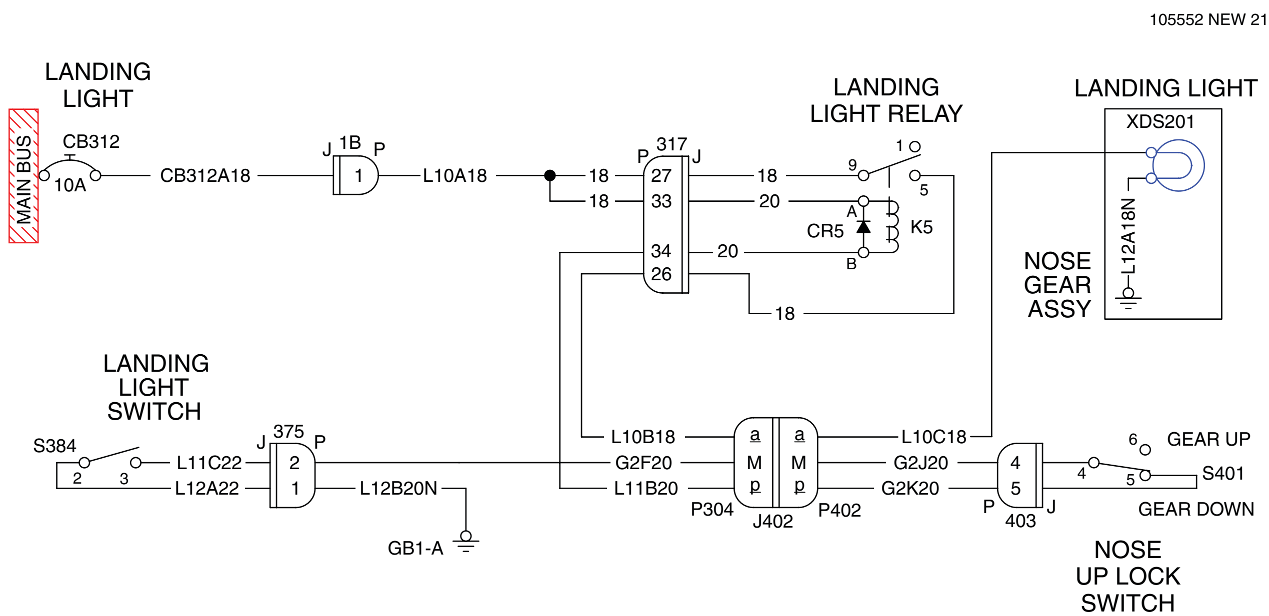
Landing and Taxi Lights
Figure 4 (Sheet 3 of 7)



Landing and Taxi Lights
Figure 4 (Sheet 4 of 7)

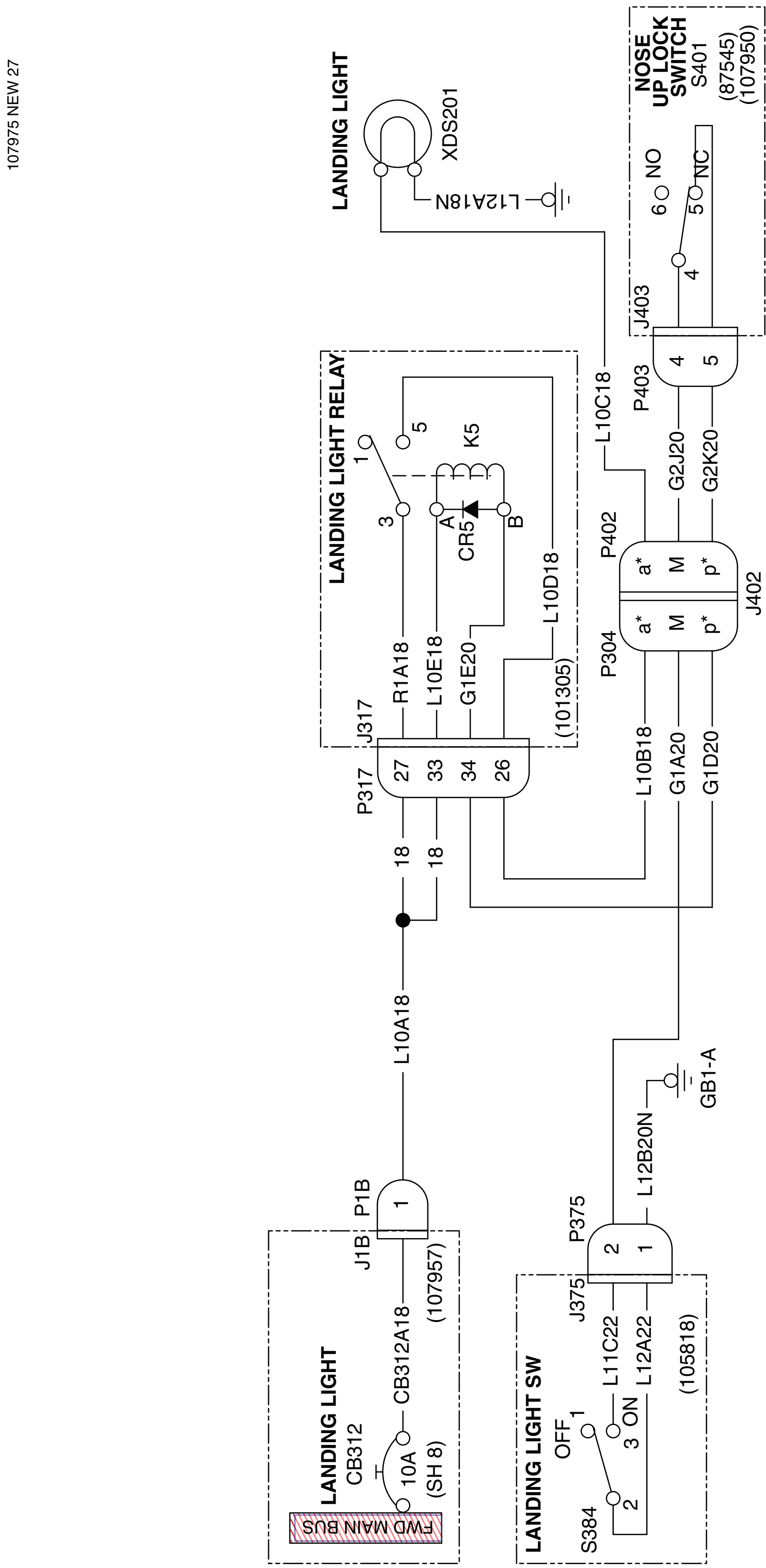


Landing and Taxi Lights
Figure 4 (Sheet 5 of 7)



SEE SH 20 FOR TAXI LIGHTS

Landing and Taxi Lights
Figure 4 (Sheet 6 of 7)



Landing and Taxi Lights
Figure 4 (Sheet 7 of 7)

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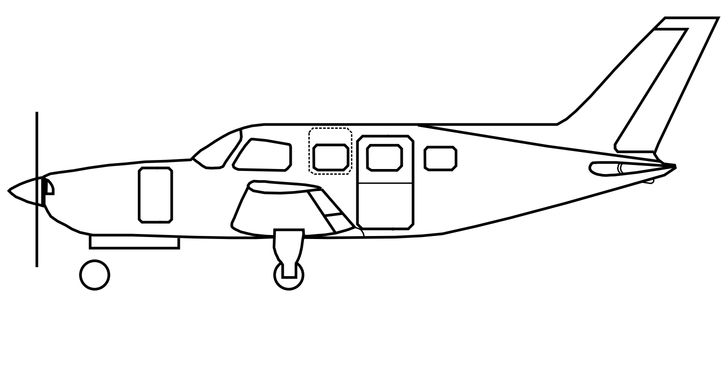
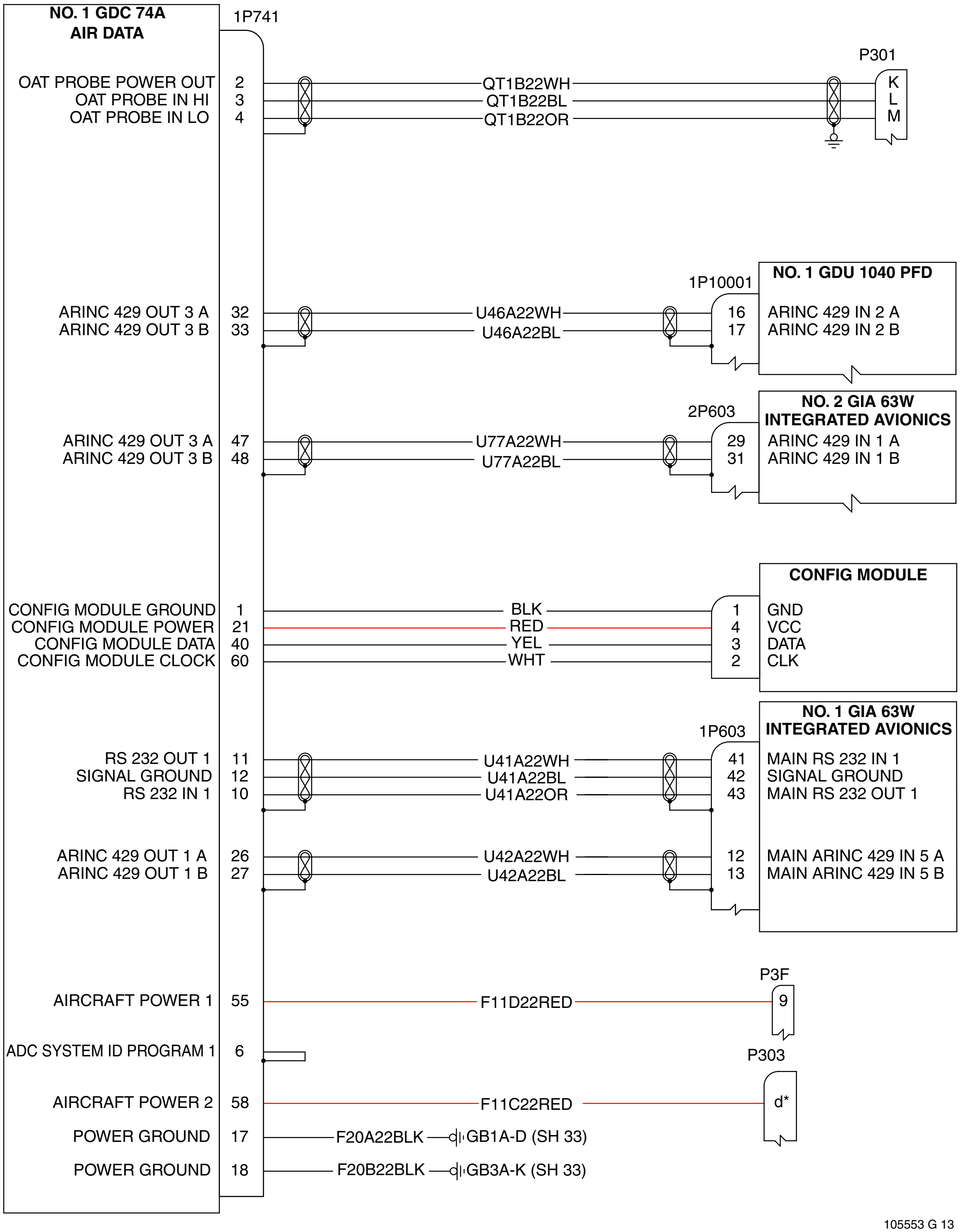
Item #	Designation	Description

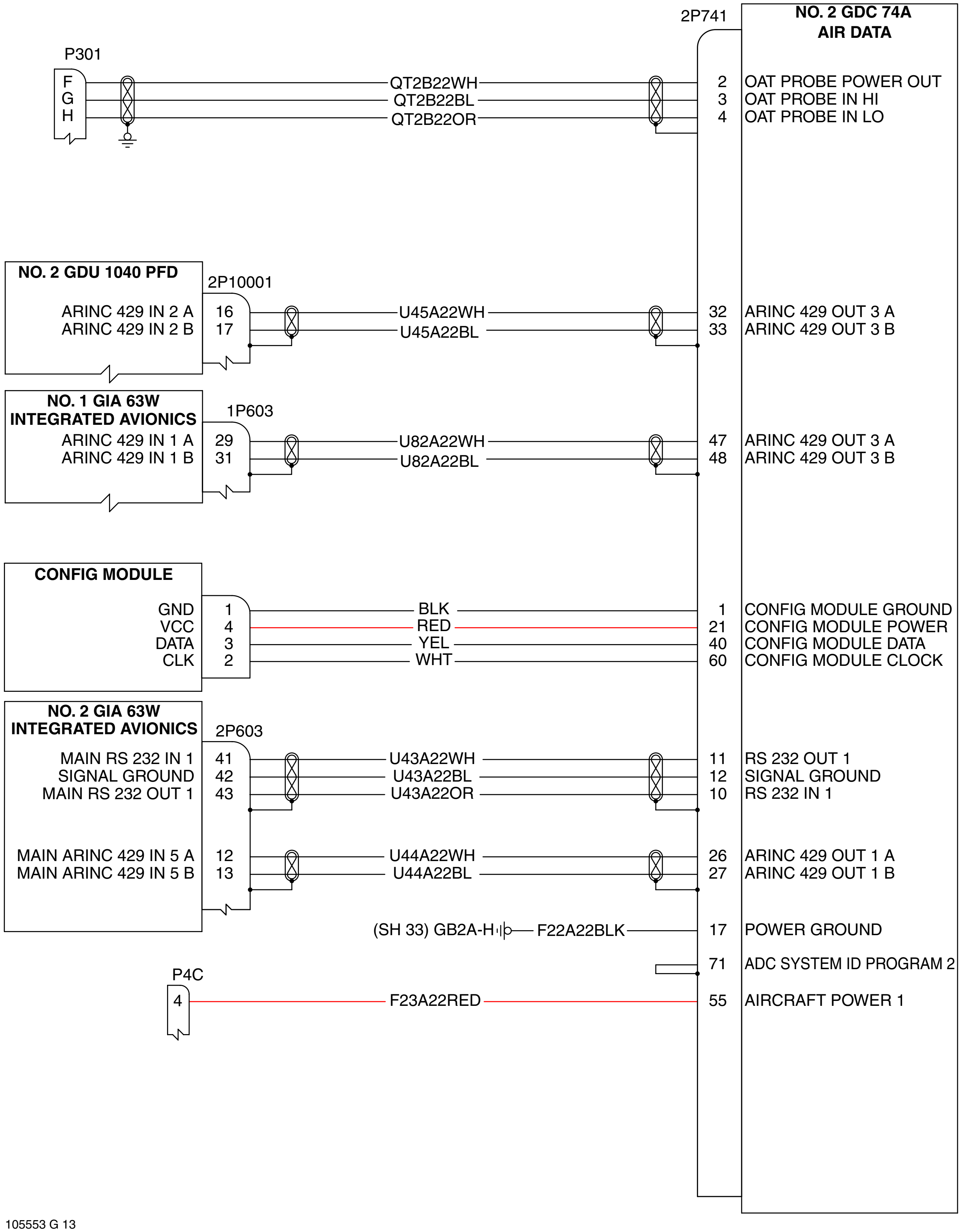
Information Pending - See Parts Catalog



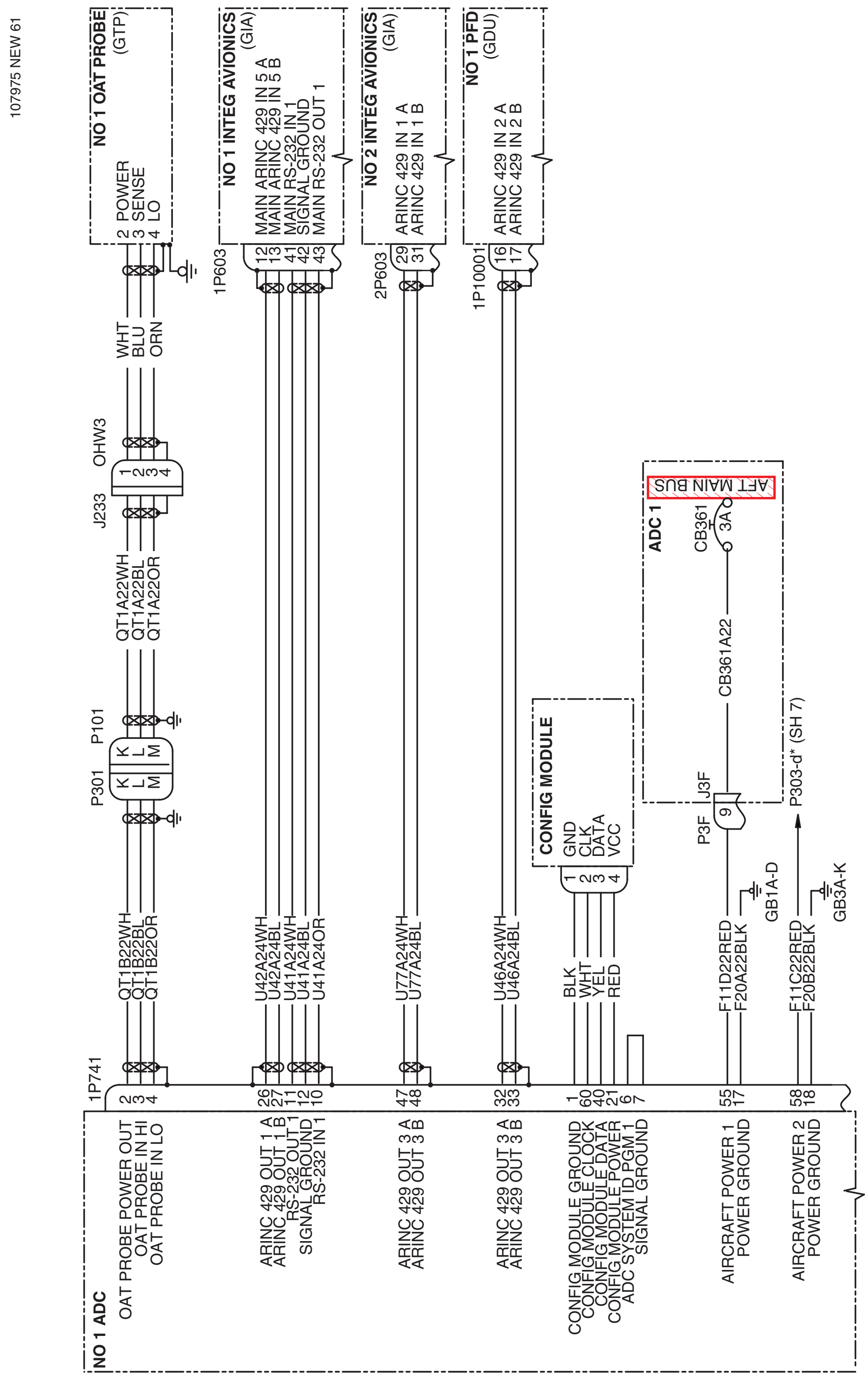
IAS - G1000 - GDC 74A
Figure 1 (Sheet 1 of 6)



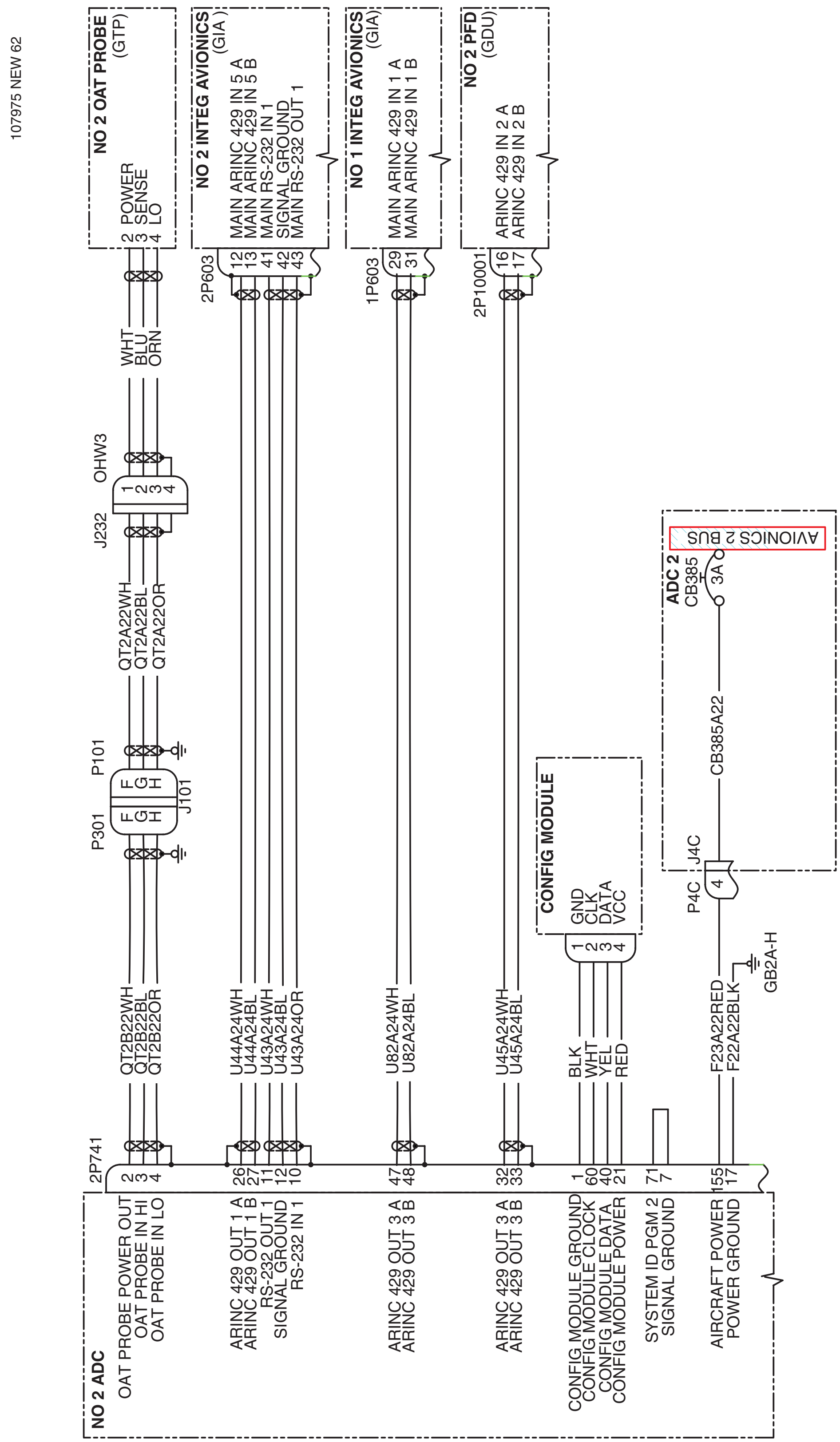
IAS - G1000 - GDC 74A
Figure 1 (Sheet 2 of 6)



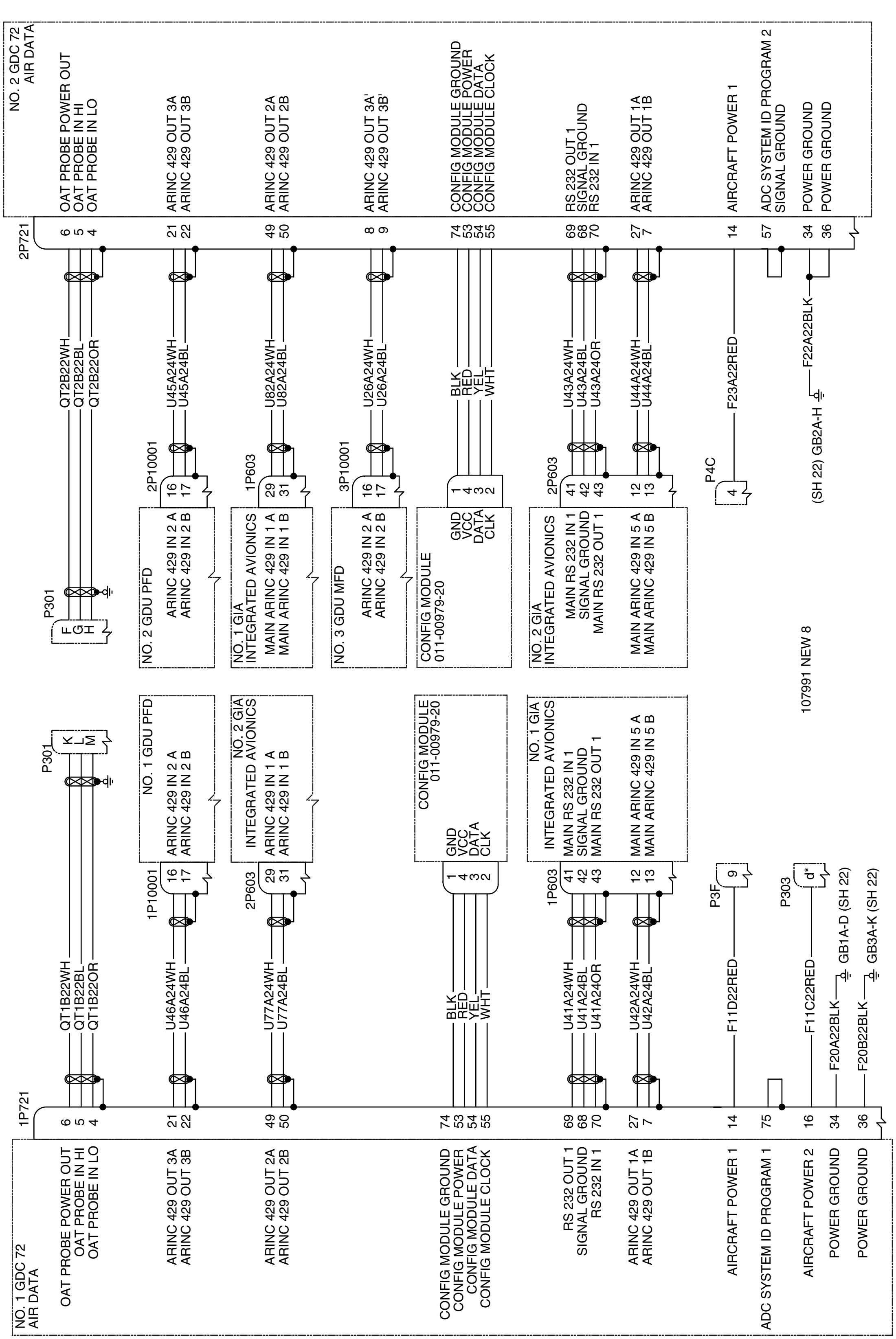
IAS - G1000 - GDC 74A
Figure 1 (Sheet 3 of 6)



IAS - G1000 - GDC 74A
Figure 1 (Sheet 4 of 6)



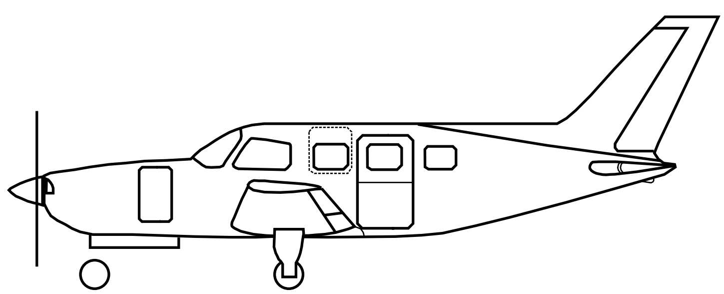
IAS - G1000 - GDC 74A
Figure 1 (Sheet 5 of 6)



IAS - G1000 - GDC 74A
Figure 1 (Sheet 6 of 6)

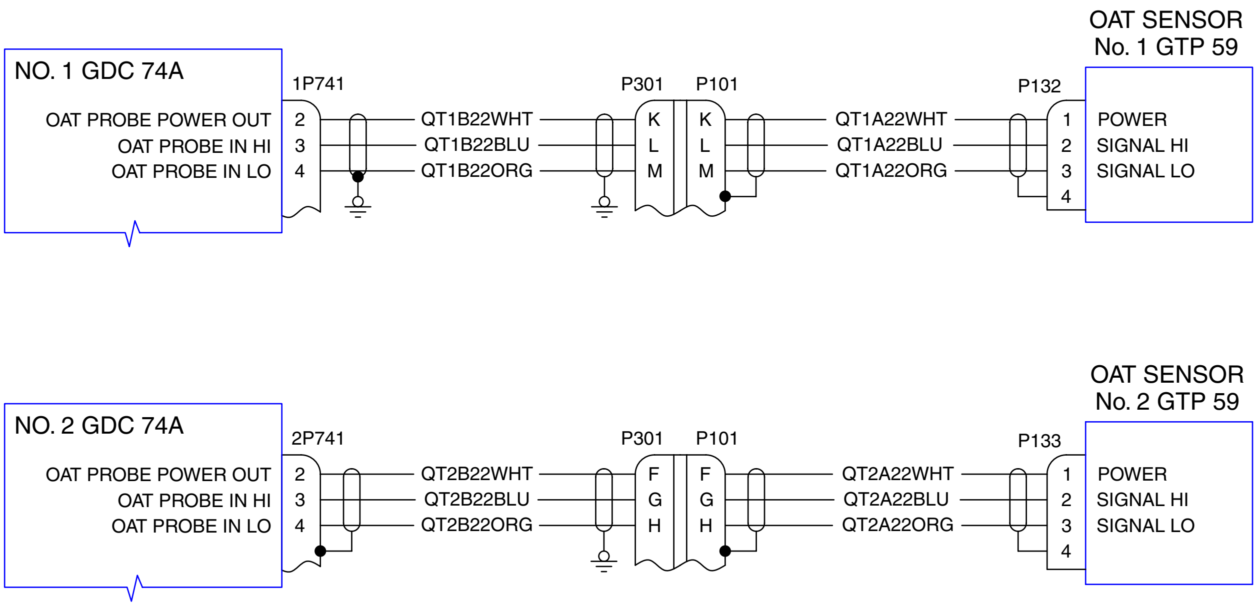
Item #	Designation	Description

Information Pending - See Parts Catalog



IAS - G1000 - OAT (GTP 59)
Figure 2 (Sheet 1 of 2)

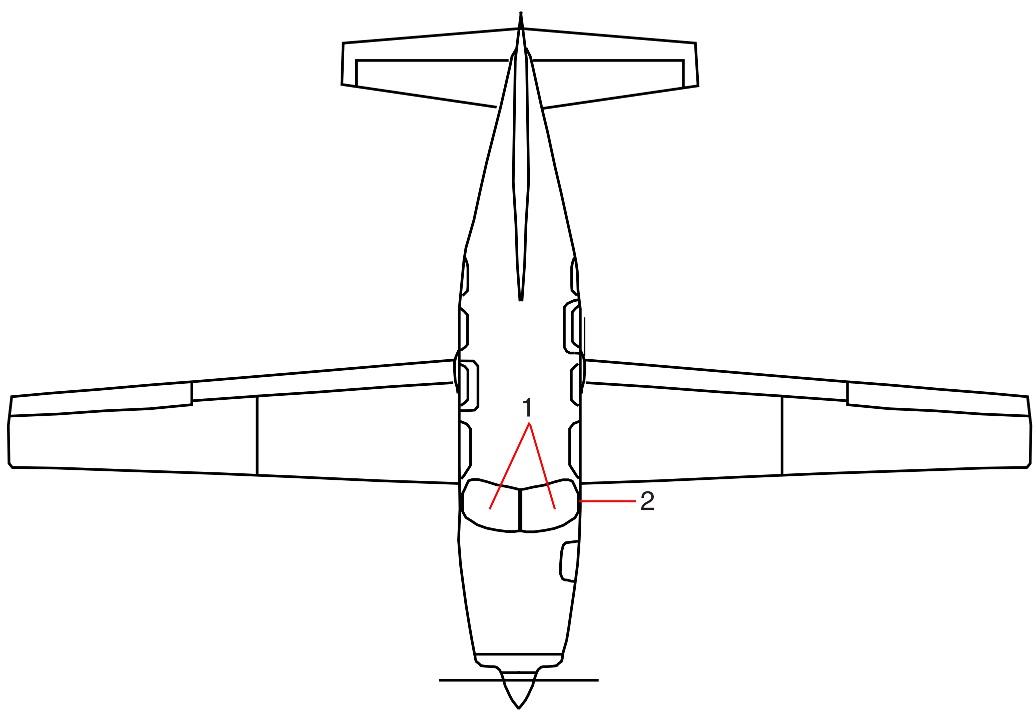
105552 NEW 37



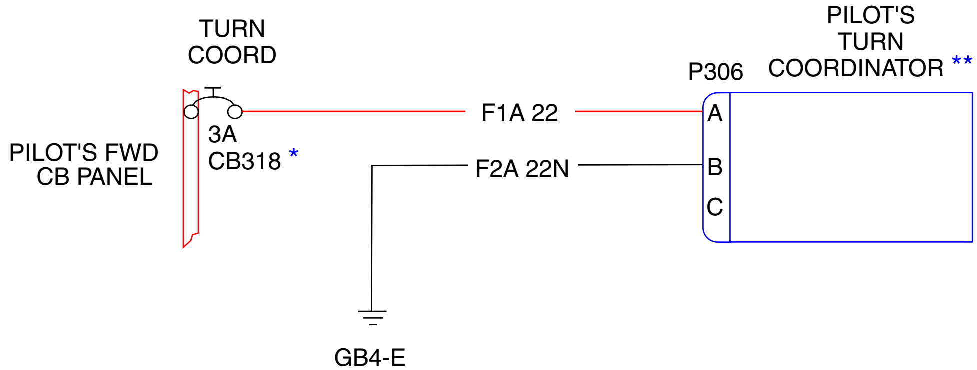
See 107975 SH 61 and SH 62 for S/N's 4636633, 4636652 and up.

IAS - G1000 - OAT (GTP 59)
Figure 2 (Sheet 2 of 2)

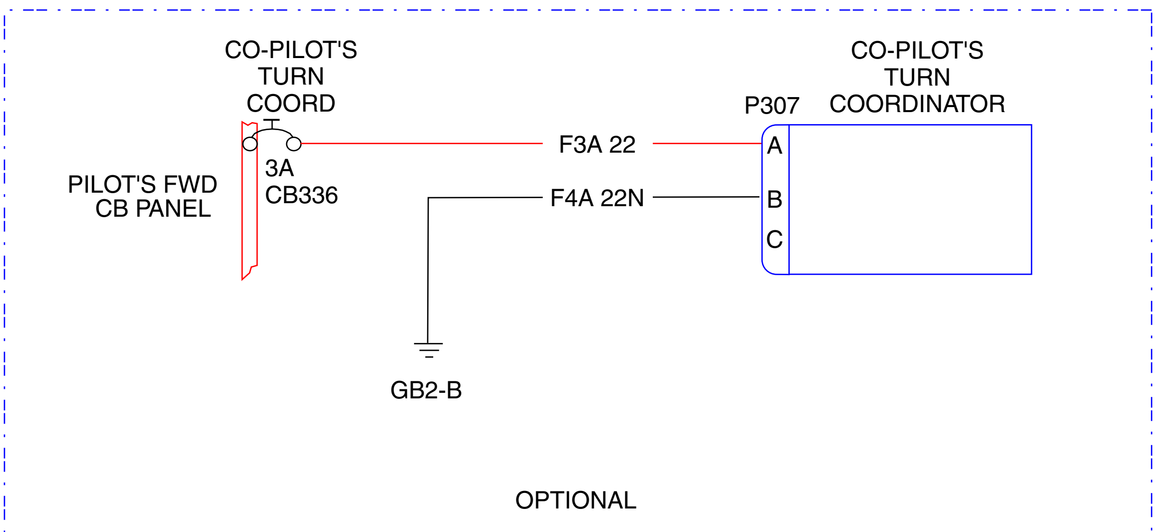
Item #	Designation	Description
1		Pilot's Turn Coordinator (Standard) Copilot's Turn Coordinator (Optional)
2	CB317, CB318, CB336	Circuit Breaker (3 Amp)



Turn Coordinator(s)
Figure 1 (Sheet 1 of 2)



* CB317 IN S/N'S 4636001 THRU 4636131 ONLY.
** IN S/N' 4636248 AND UP, SEE AUTOPILOT C.I.P. FOR PILOT'S TURN COORDINATOR WIRING.

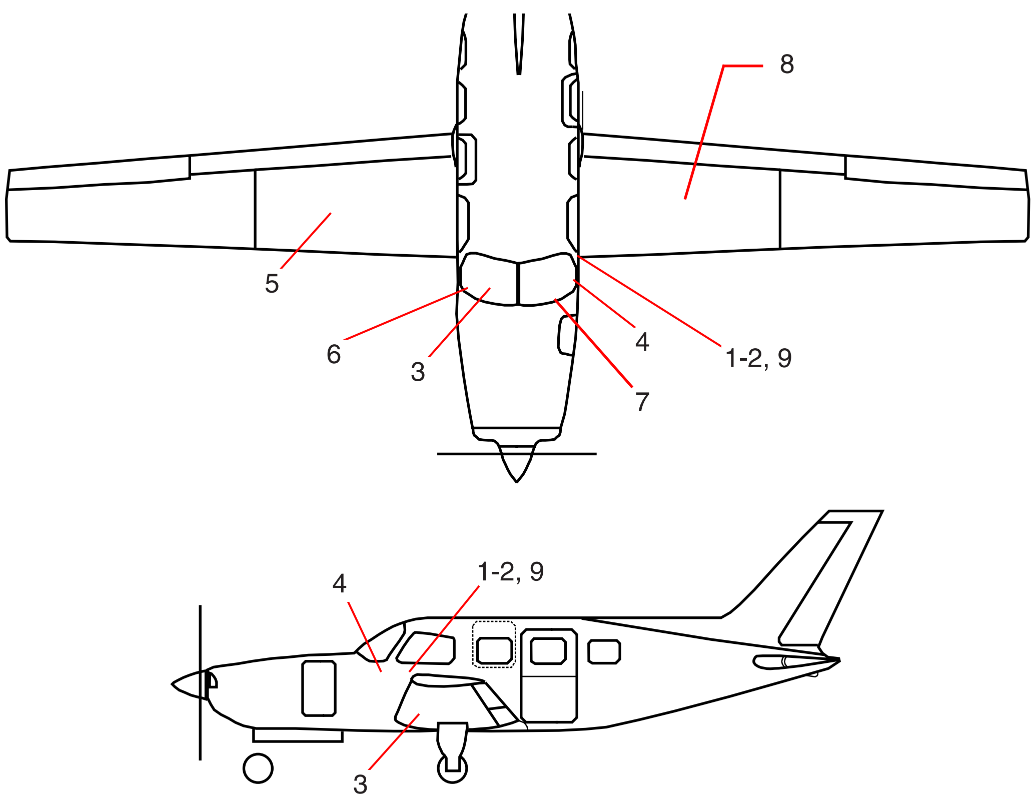


89801 AH 38.0
85508 L
101238 E
101301 C
104051 C
104301 NEW/S

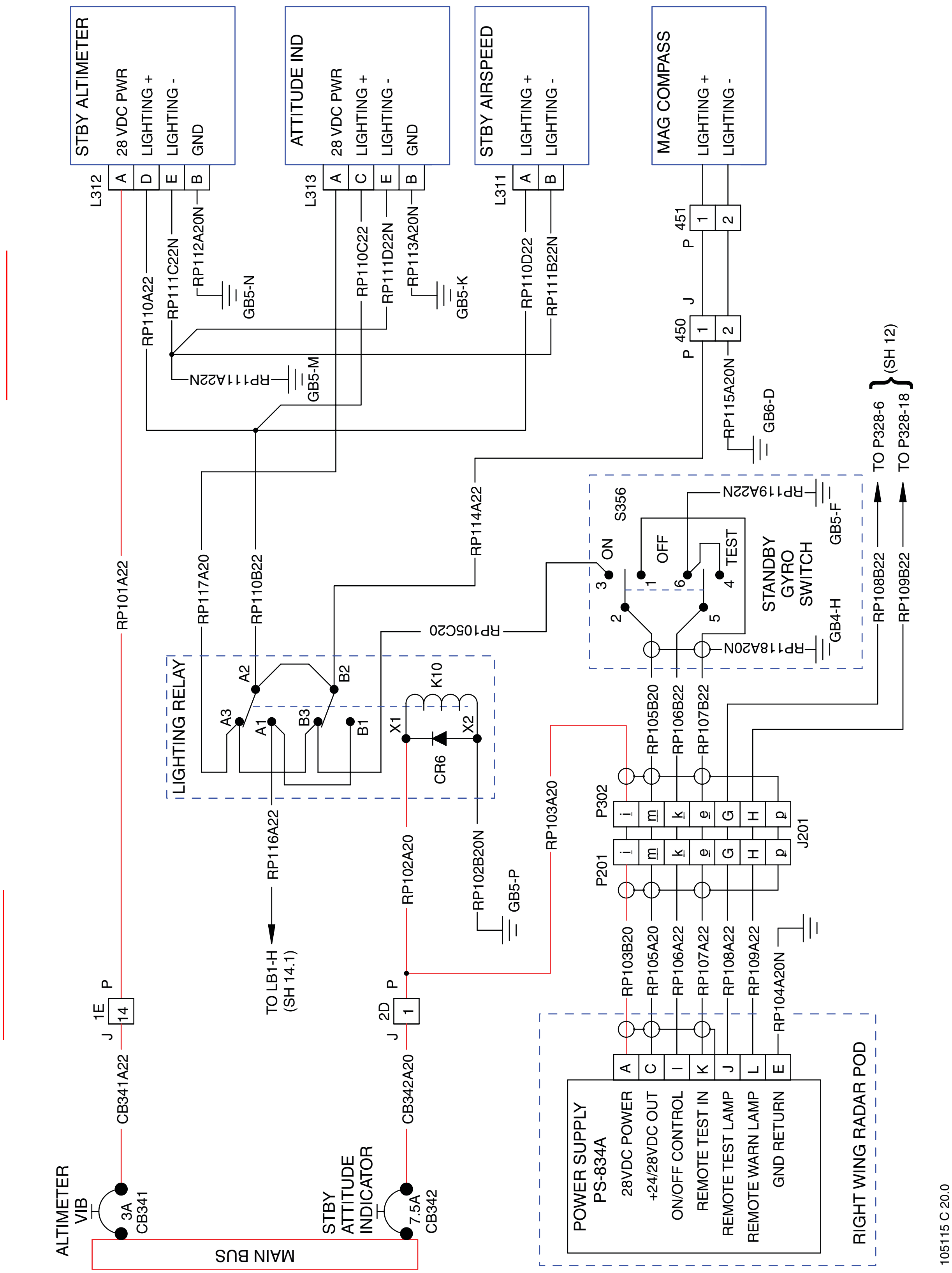
Turn Coordinator(s)
Figure 1 (Sheet 2 of 2)

Item #	Designation	Description
1	CB341	Circuit Breaker - Altimeter (3 Amp)
2	CB342	Circuit Breaker - Standby Attitude Indicator (7.5 Amp)
3	K-1	Lighting Relay
4	S356	Standby Gyro Switch
5	PS-834A	Power Supply
6	CB30	Circuit Breaker - Standby Attitude Indicator (1 Amp)
7[1]	EBD	Aspen Avionics Evolution Backup Display
8[1]	RSM	Remote Sensor Module
9[1]	CB335	Circuit Breaker - Standby Attitude Indicator (7.5 Amp)

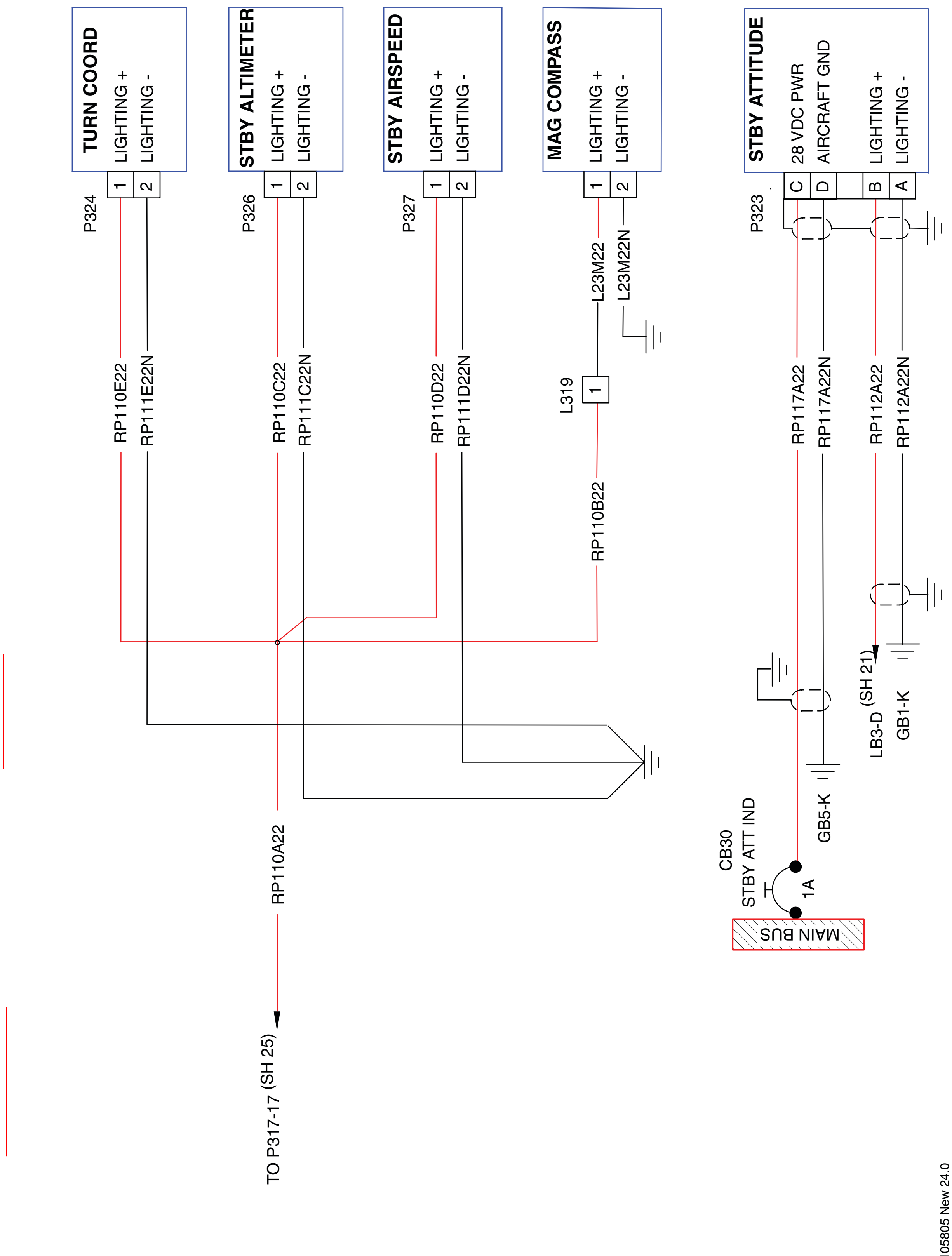
(1) 4636633, 4636652–4636775.



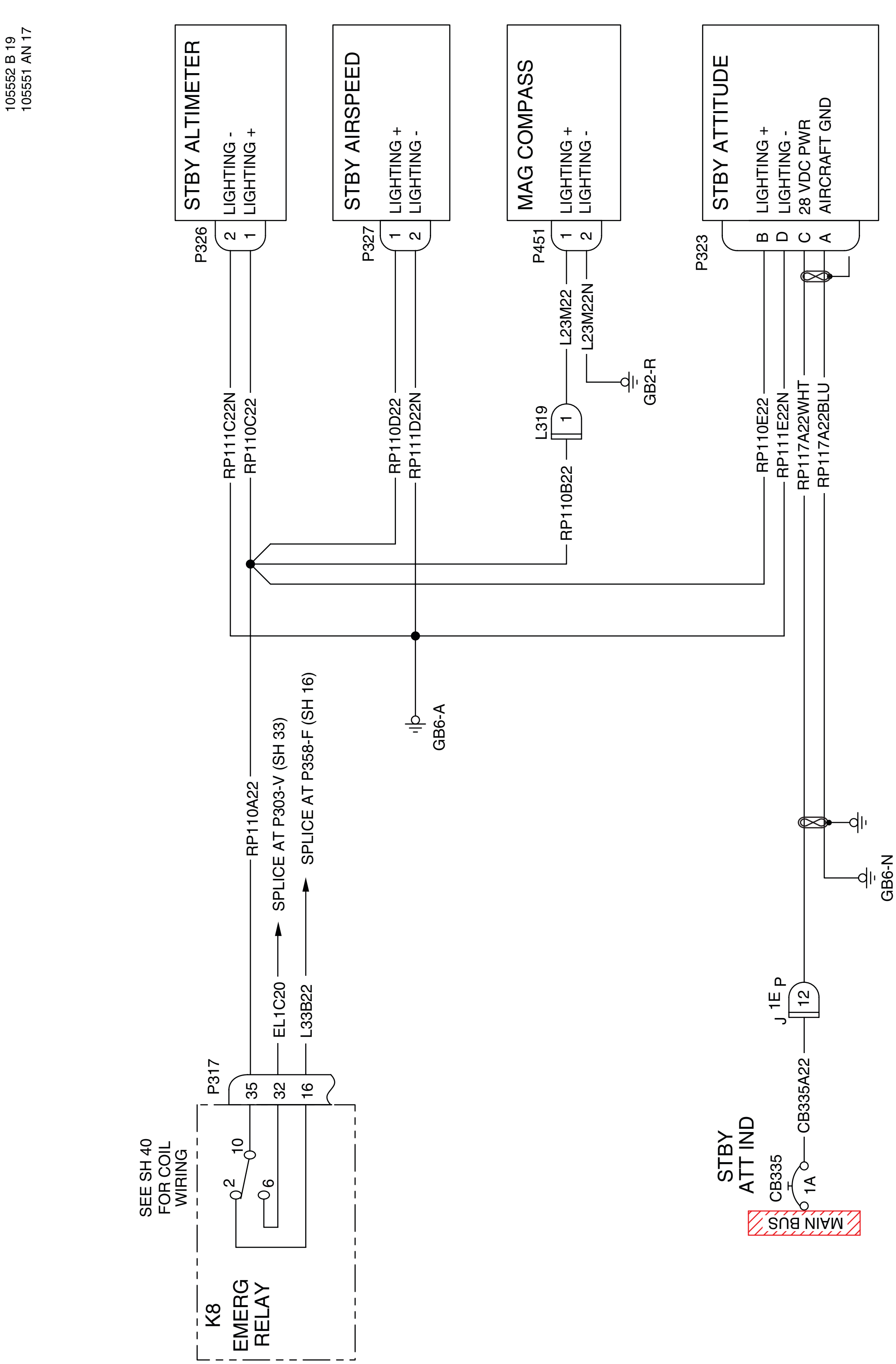
Standby Instruments
Figure 2 (Sheet 1 of 5)



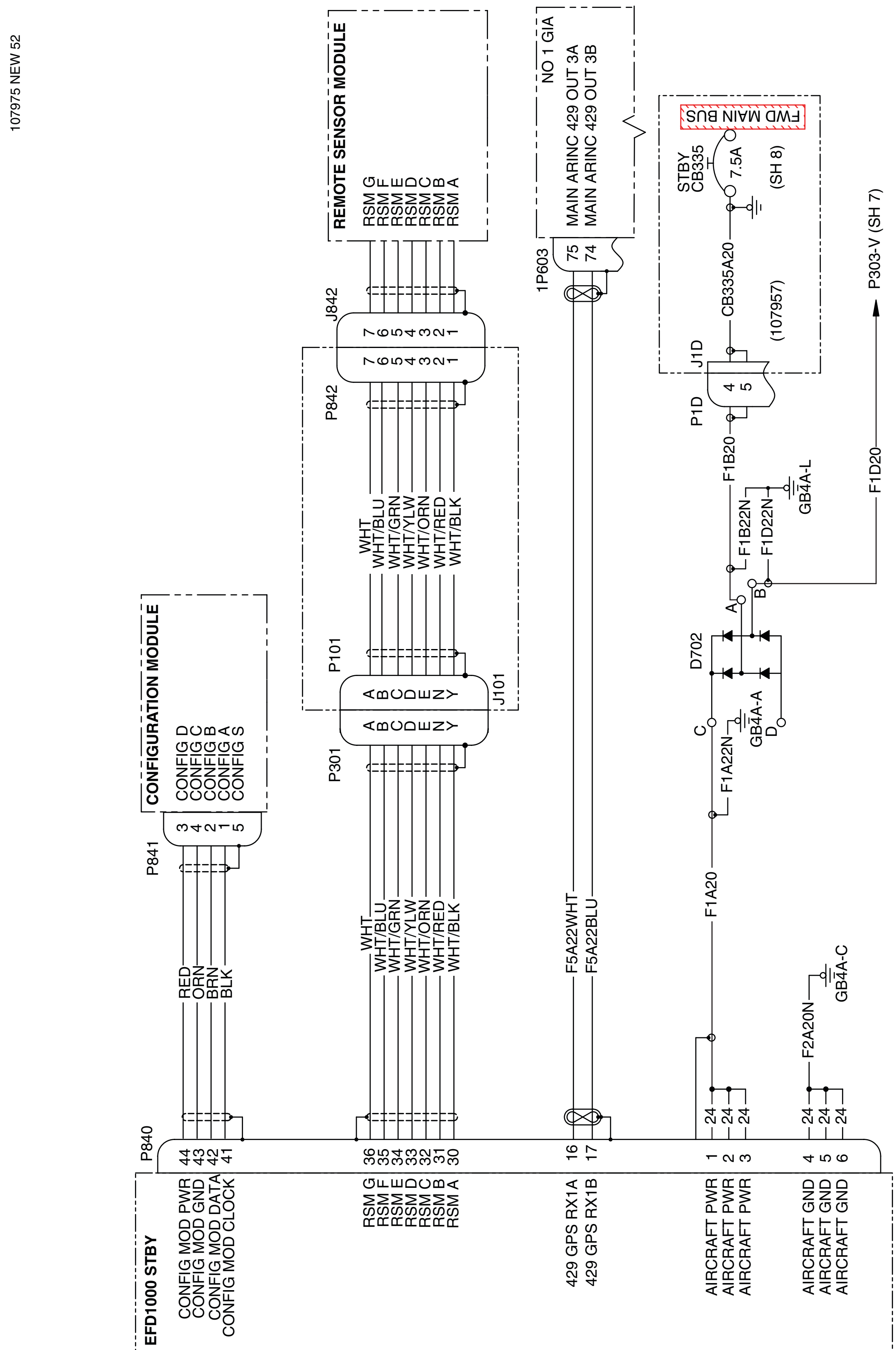
Standby Instruments
Figure 2 (Sheet 2 of 5)



Standby Instruments
Figure 2 (Sheet 3 of 5)



Standby Instruments
Figure 2 (Sheet 4 of 5)



Standby Instruments
Figure 2 (Sheet 5 of 5)

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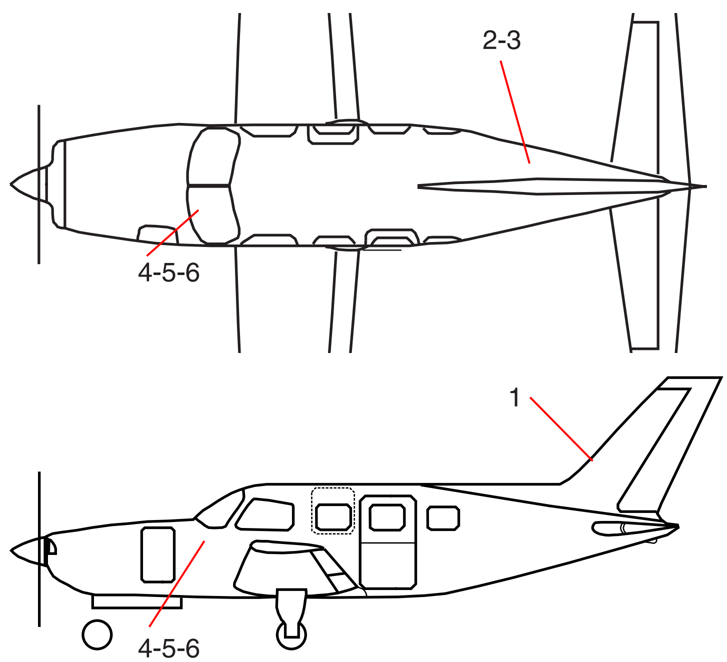
Item #	Designation	Description
1	S-TEC 6446	Flux Sensor
2	ADAHRS Select	Relay
3	Meggitt 37002-01-02	ADAHRS Unit
4	ADAHRS Select	Switch
5	Meggitt 84-133-5	PFD Unit
6	Meggitt 84-134-5	ND Unit



Electronic Flight Instrument System (EFIS) - Meggitt (Optional)
Figure 3 (Sheet 1 of 5)

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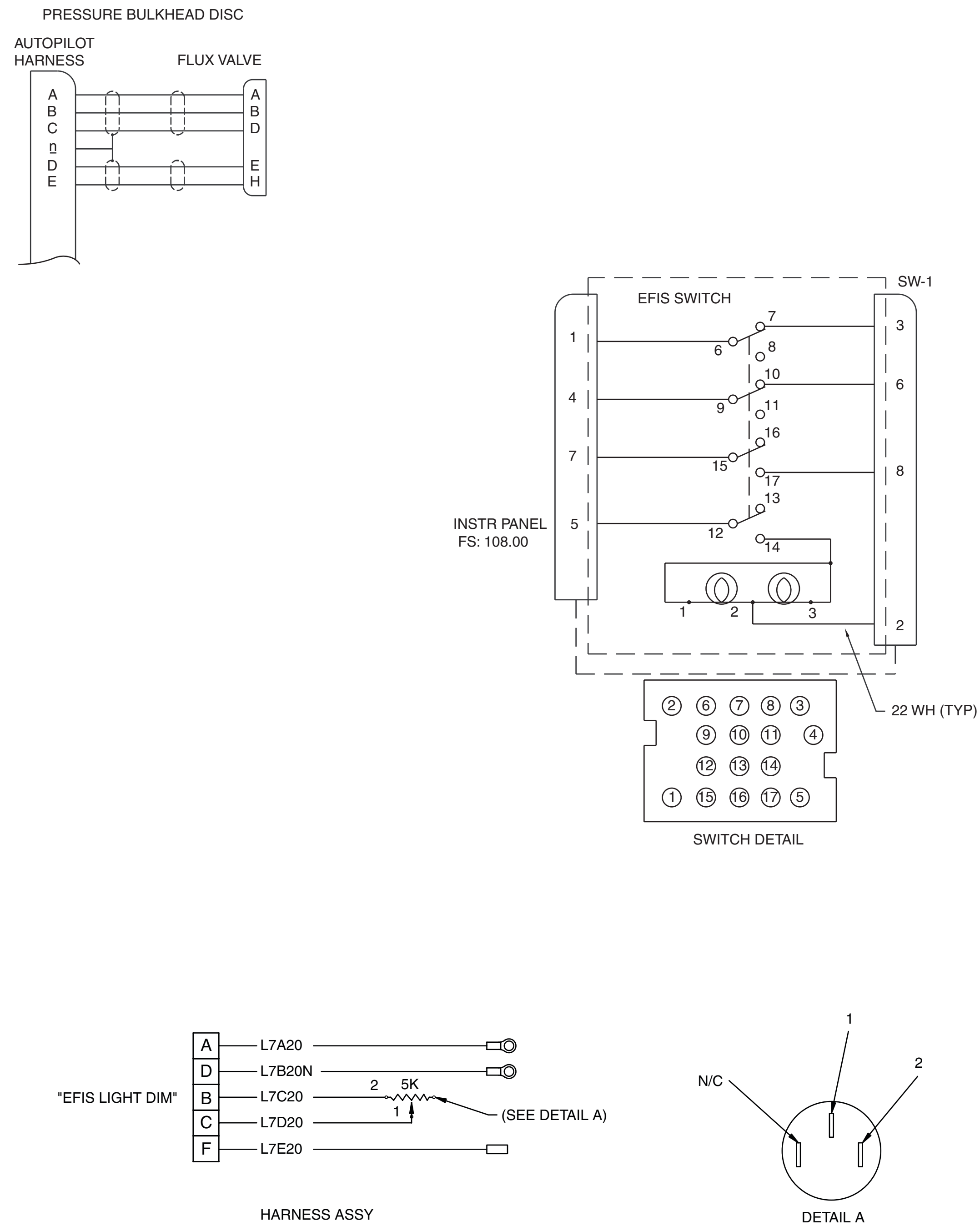
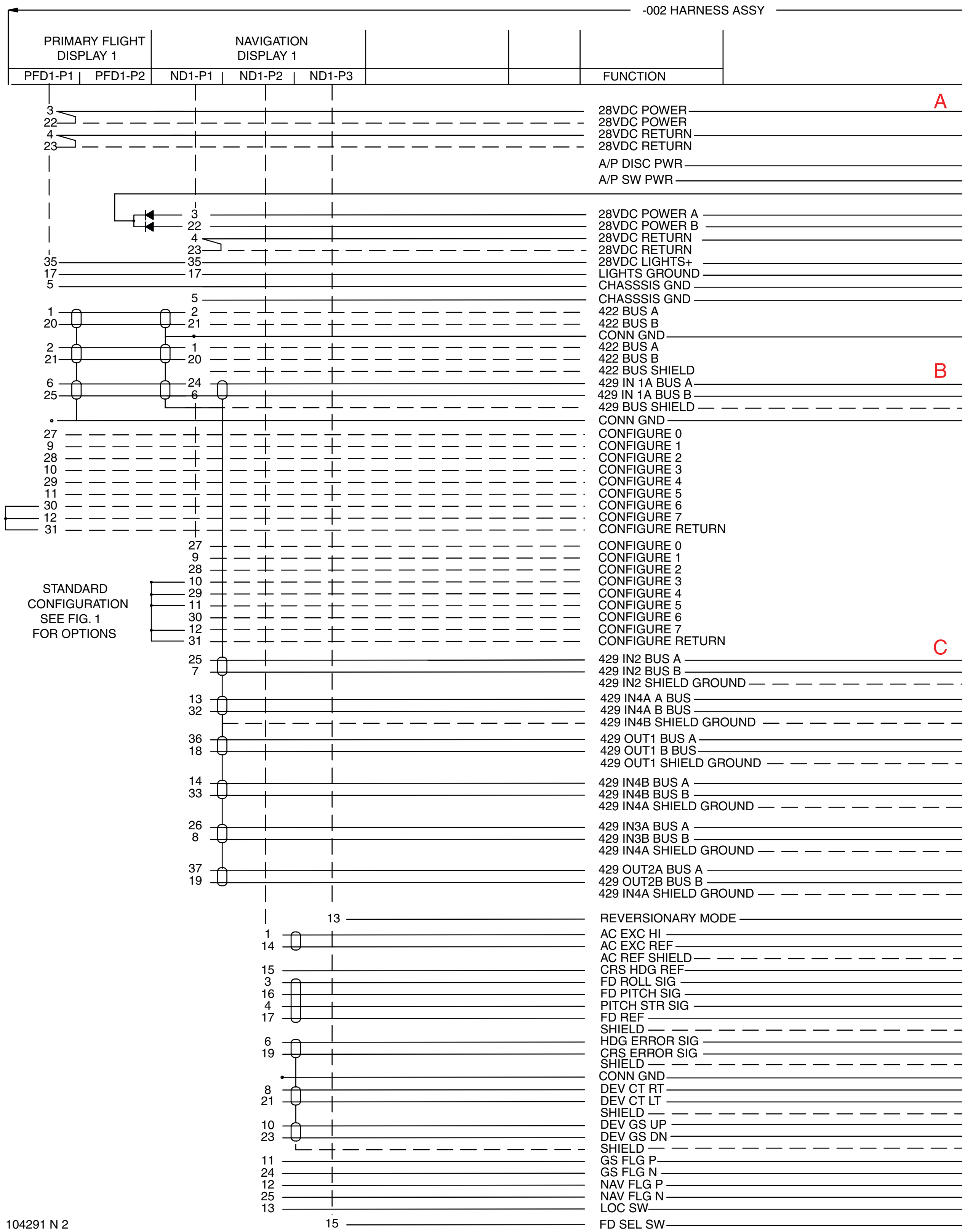
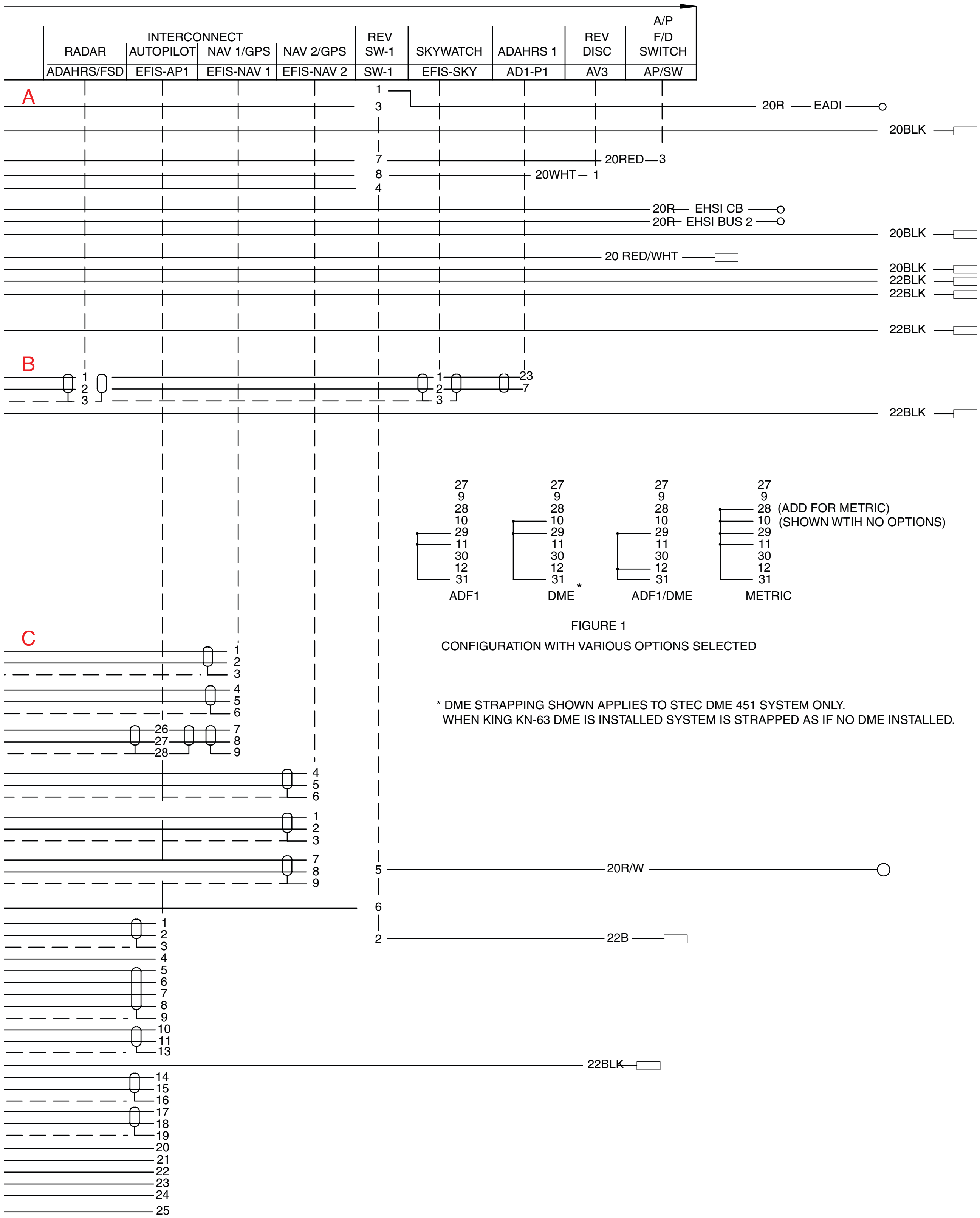


104291 M

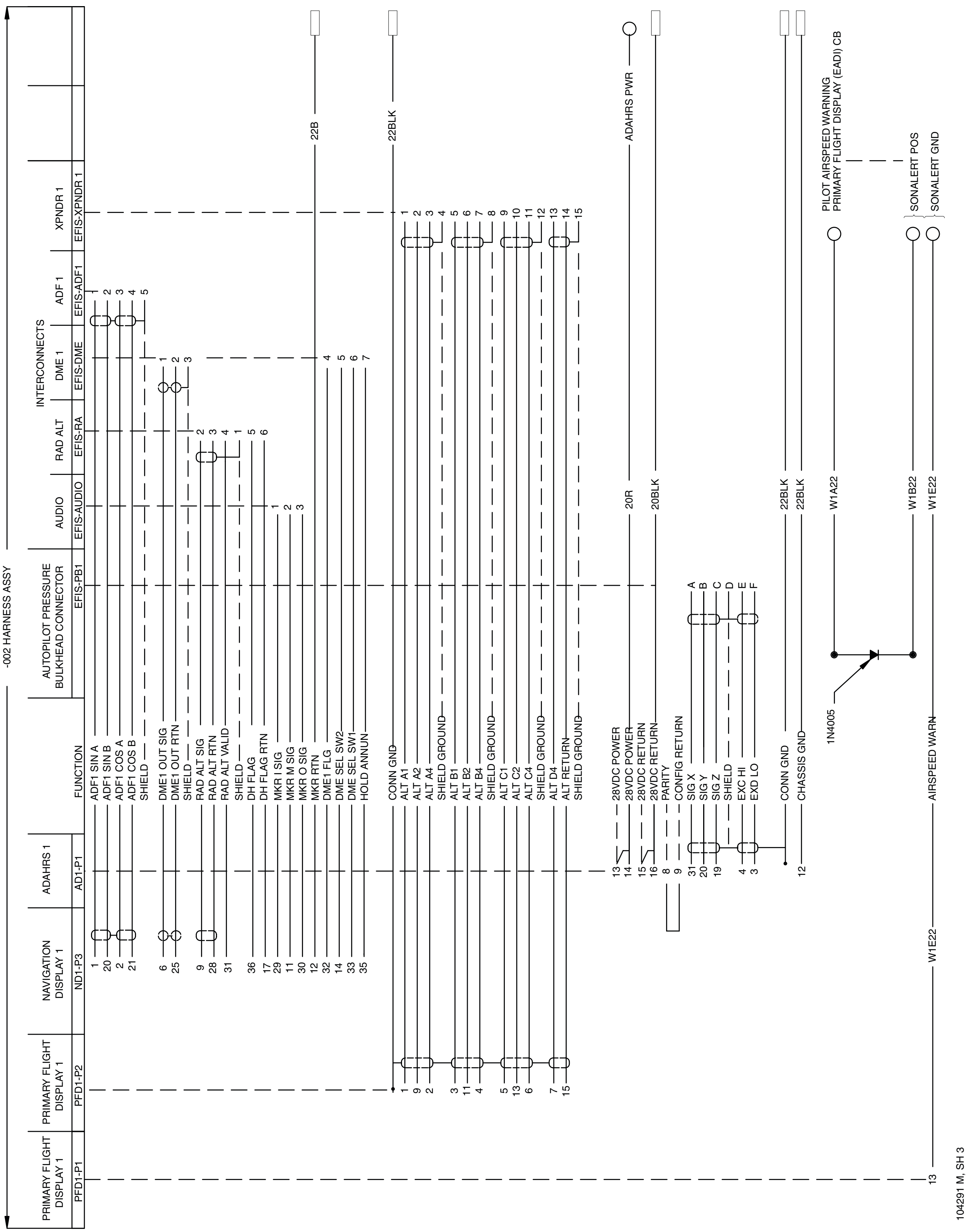
Electronic Flight Instrument System (EFIS) - Meggitt (Optional)
Figure 3 (Sheet 2 of 5)



Electronic Flight Instrument System (EFIS) - Meggitt (Optional)
Figure 3 (Sheet 3 of 5)



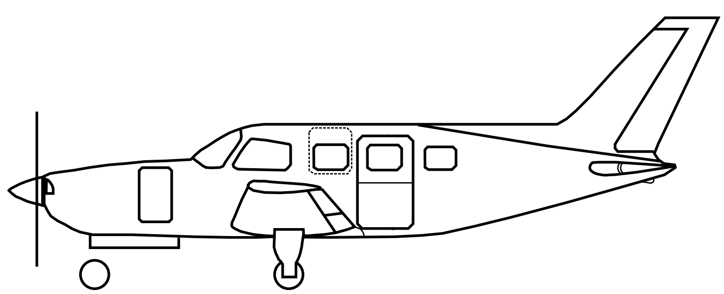
Electronic Flight Instrument System (EFIS) - Meggitt (Optional)
Figure 3 (Sheet 4 of 5)



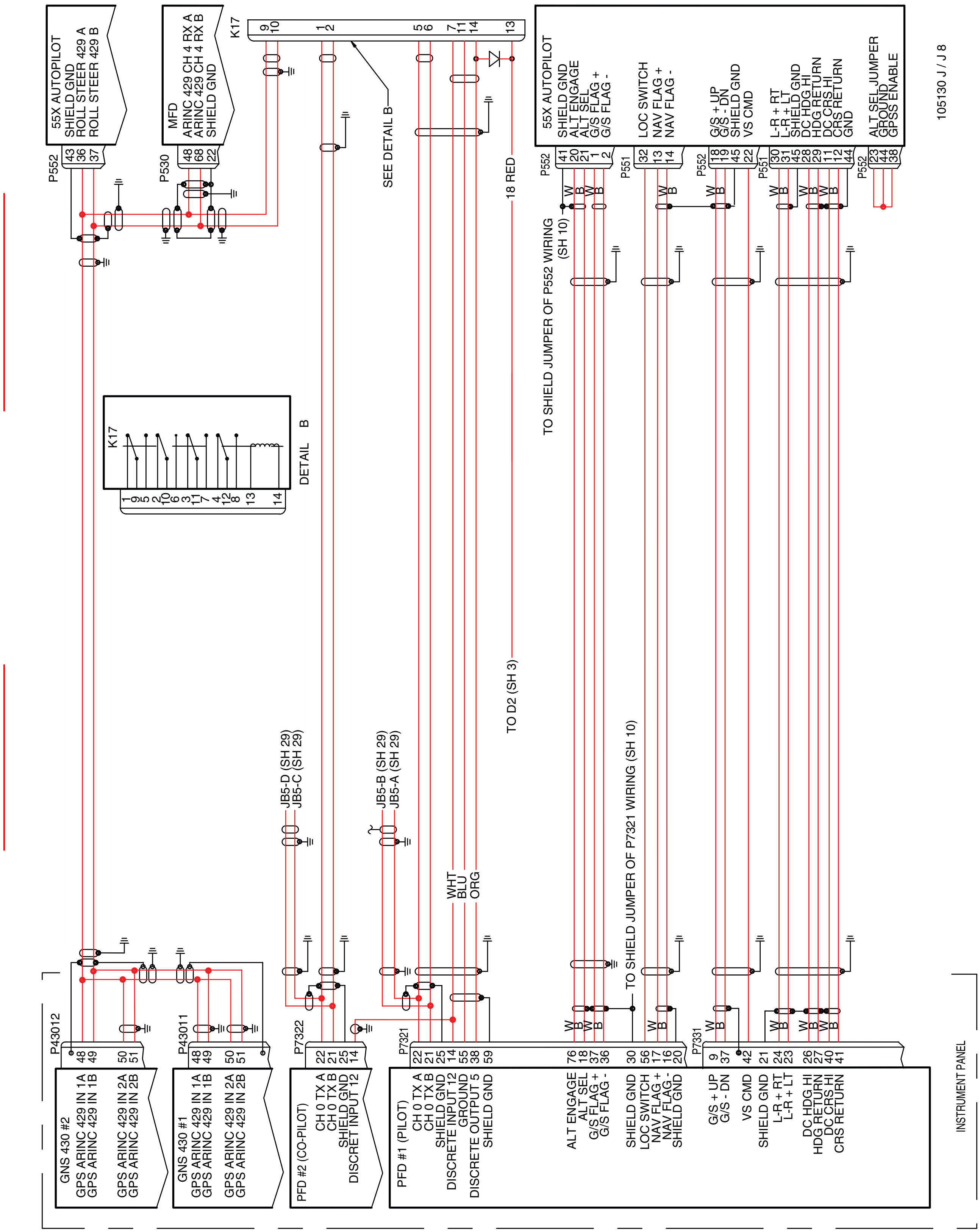
Electronic Flight Instrument System (EFIS) - Meggitt (Optional)
Figure 3 (Sheet 5 of 5)

Item #	Designation	Description

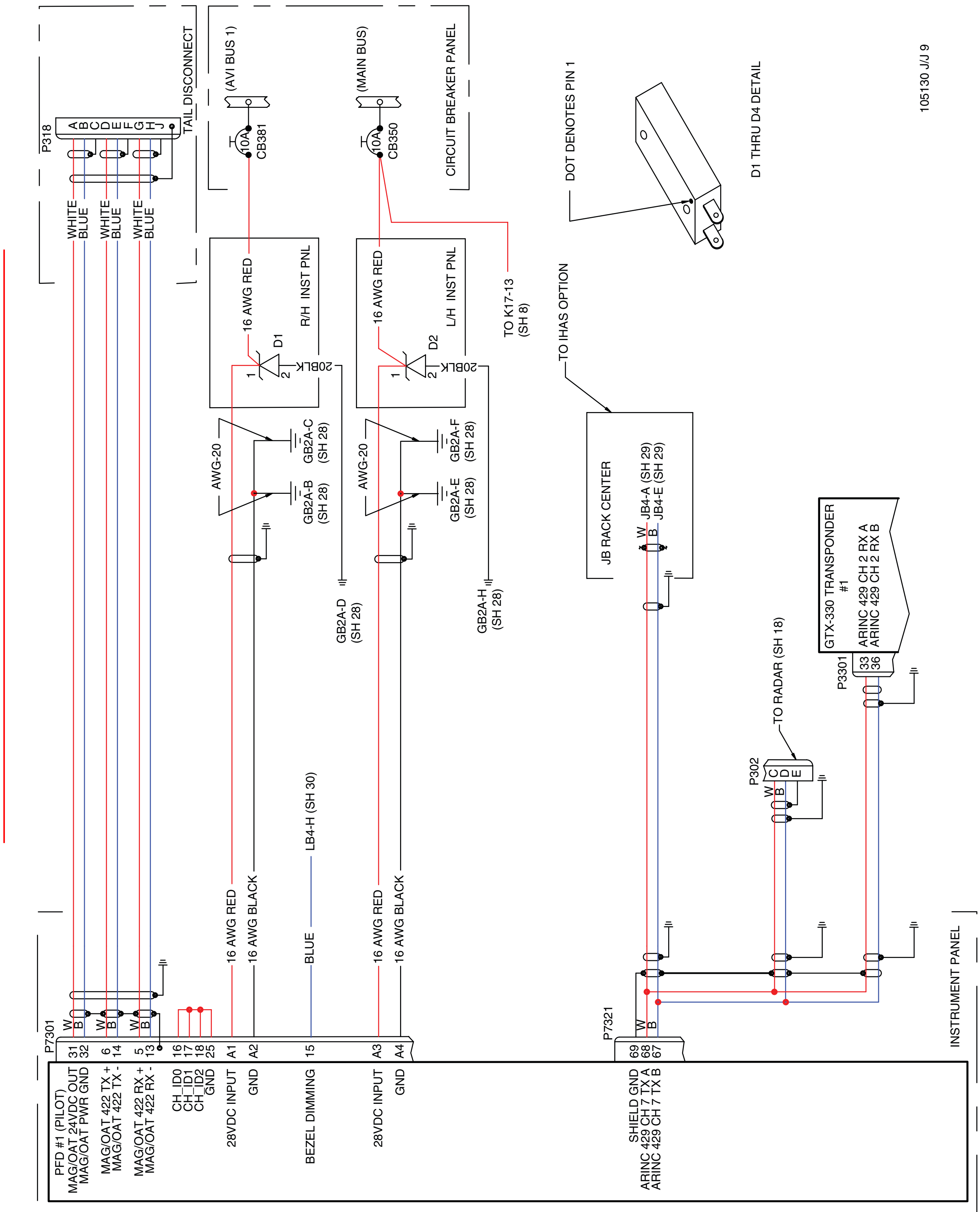
Information Pending - See Parts Catalog



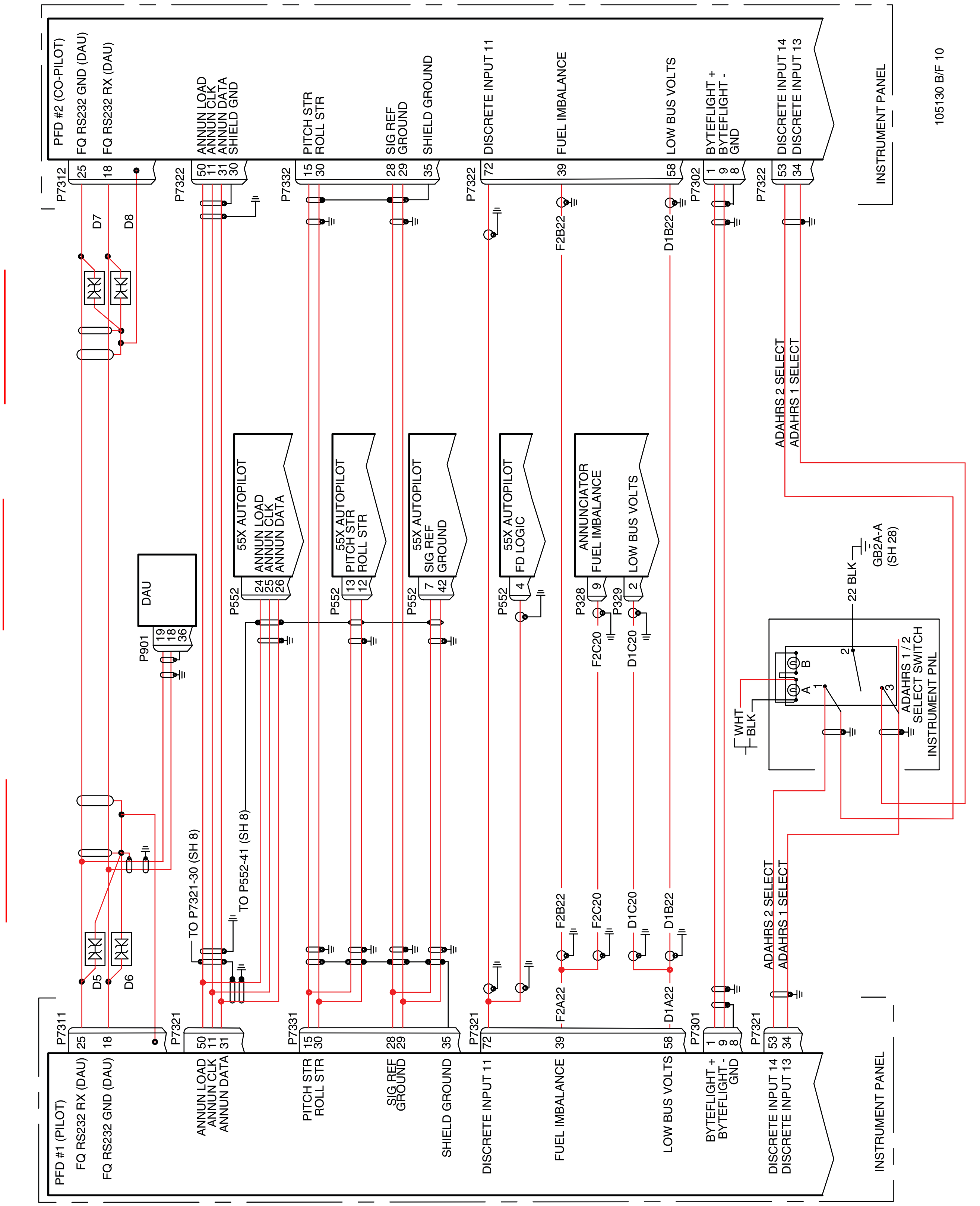
EFIS - Avidyne Entegra - Dual PFD
Figure 4 (Sheet 1 of 5)



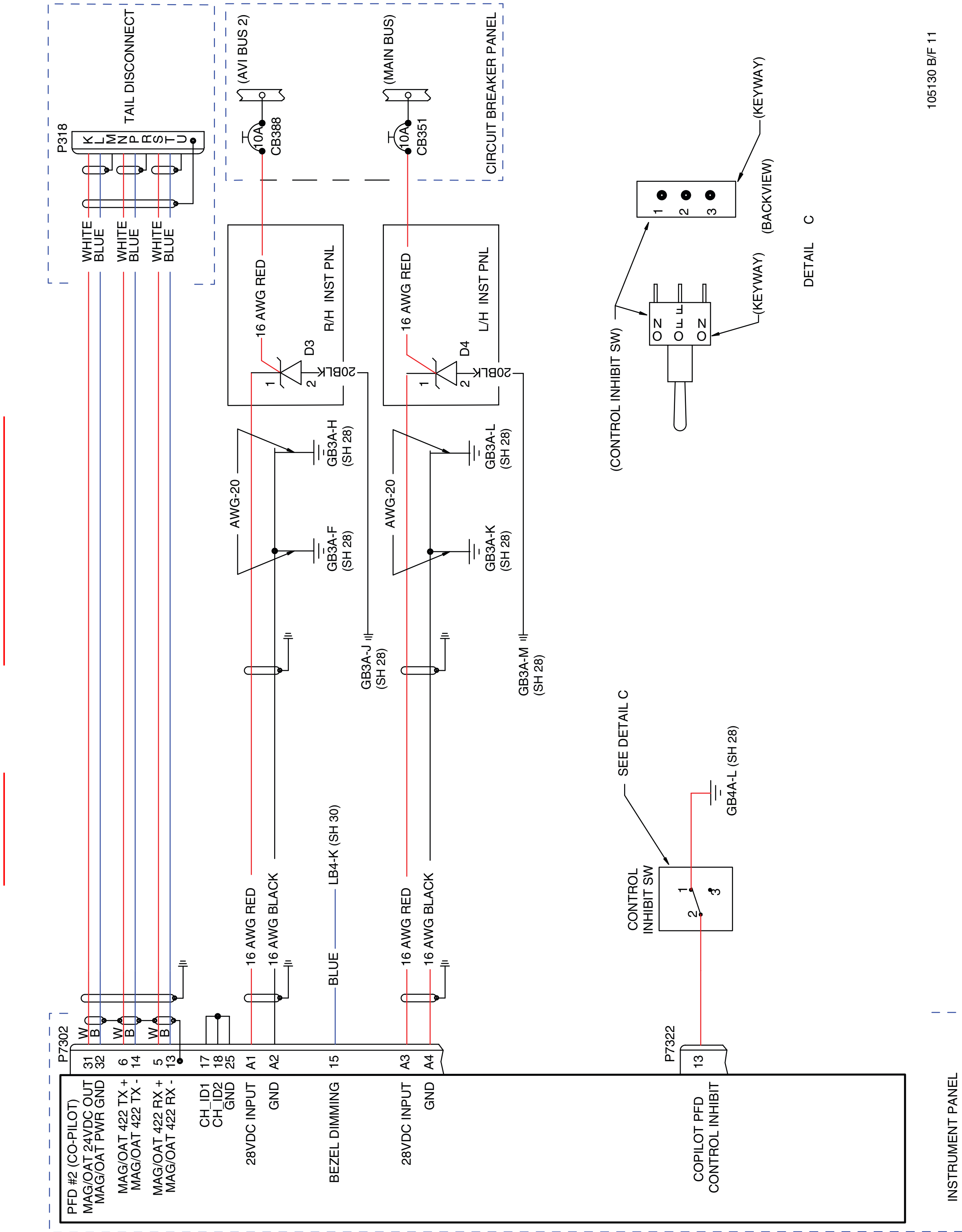
EFIS - Avidyne Entegra - Dual PFD
Figure 4 (Sheet 2 of 5)



EFIS - Avidyne Entegra - Dual PFD
Figure 4 (Sheet 3 of 5)



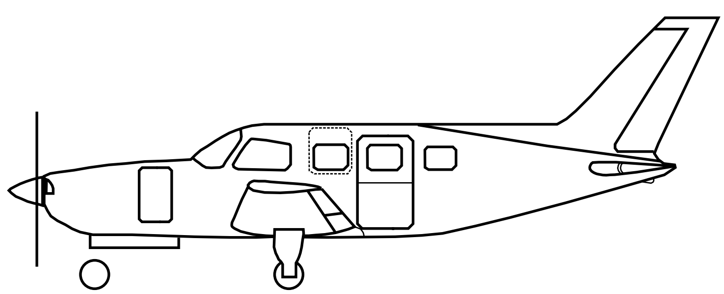
EFIS - Avidyne Entegra - Dual PFD
Figure 4 (Sheet 4 of 5)



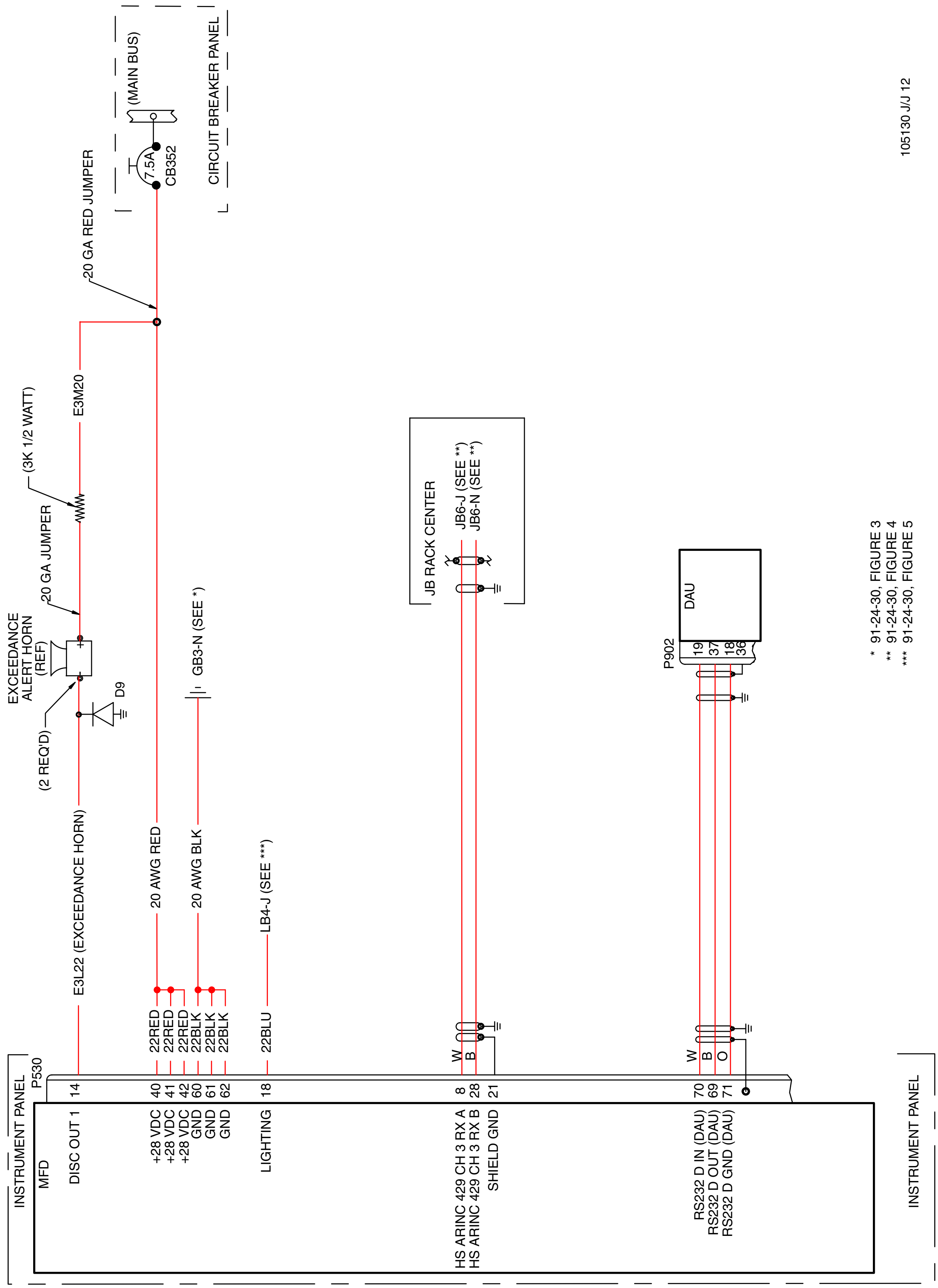
EFIS - Avidyne Entegra - Dual PFD
Figure 4 (Sheet 5 of 5)

Item #	Designation	Description

Information Pending - See Parts Catalog



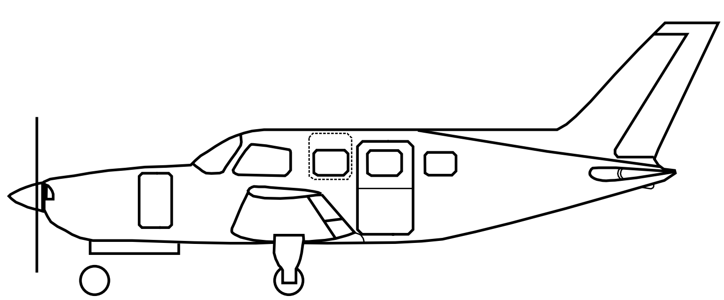
EFIS - Avidyne Entegra - MFD with Dual PFDs
Figure 5 (Sheet 1 of 2)



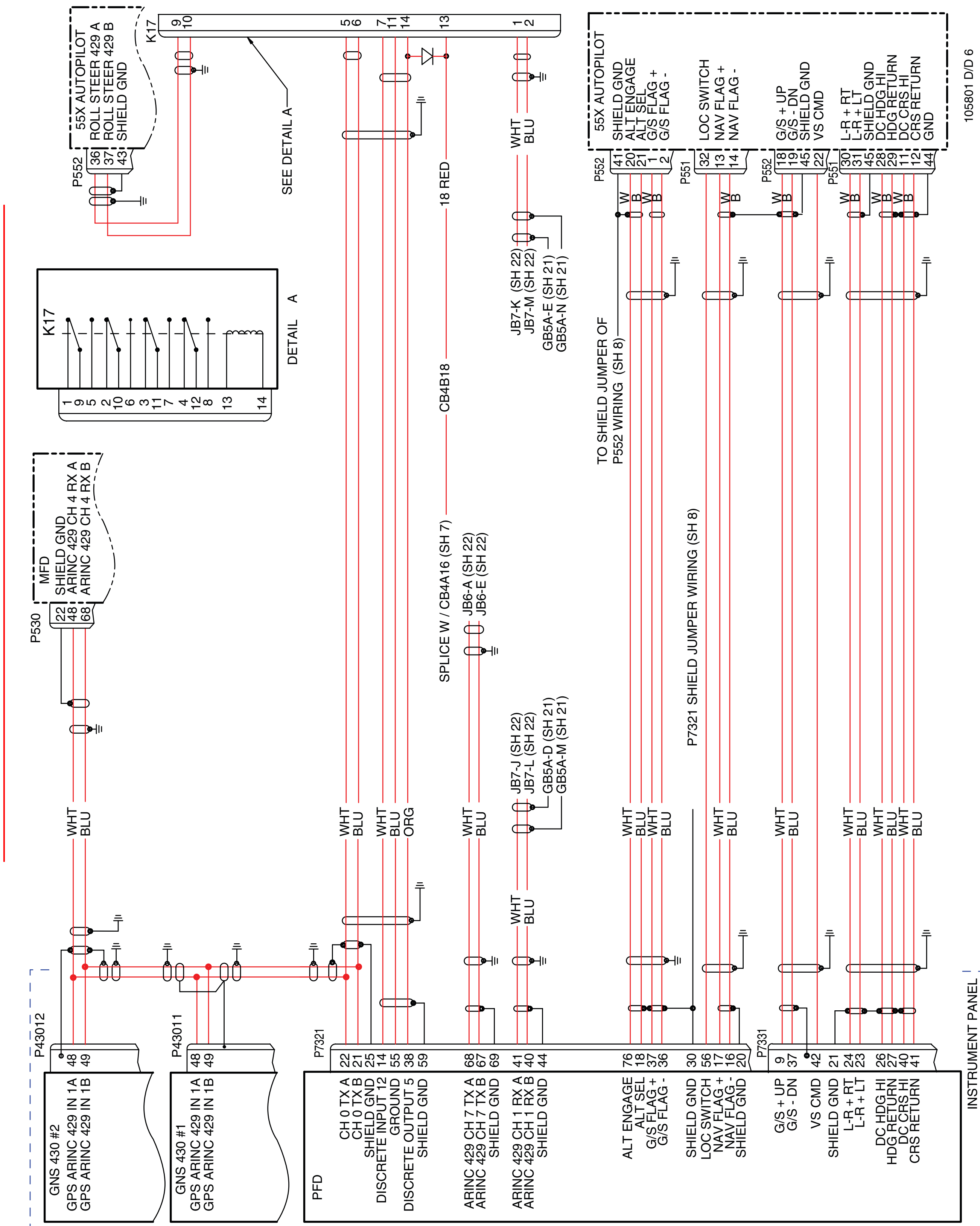
EFIS - Avidyne Entegra - MFD with Dual PFDs
Figure 5 (Sheet 2 of 2)

Item #	Designation	Description

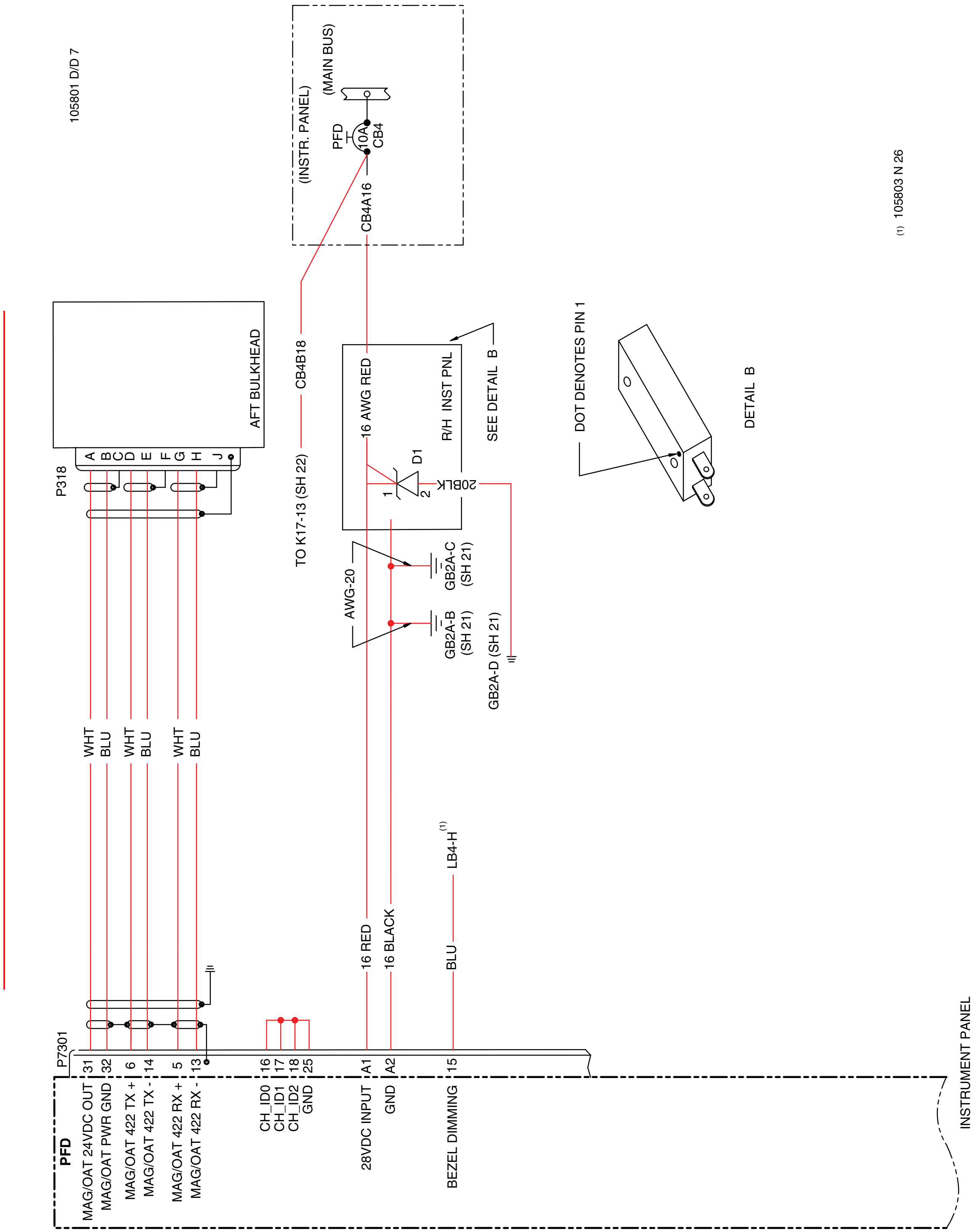
Information Pending - See Parts Catalog



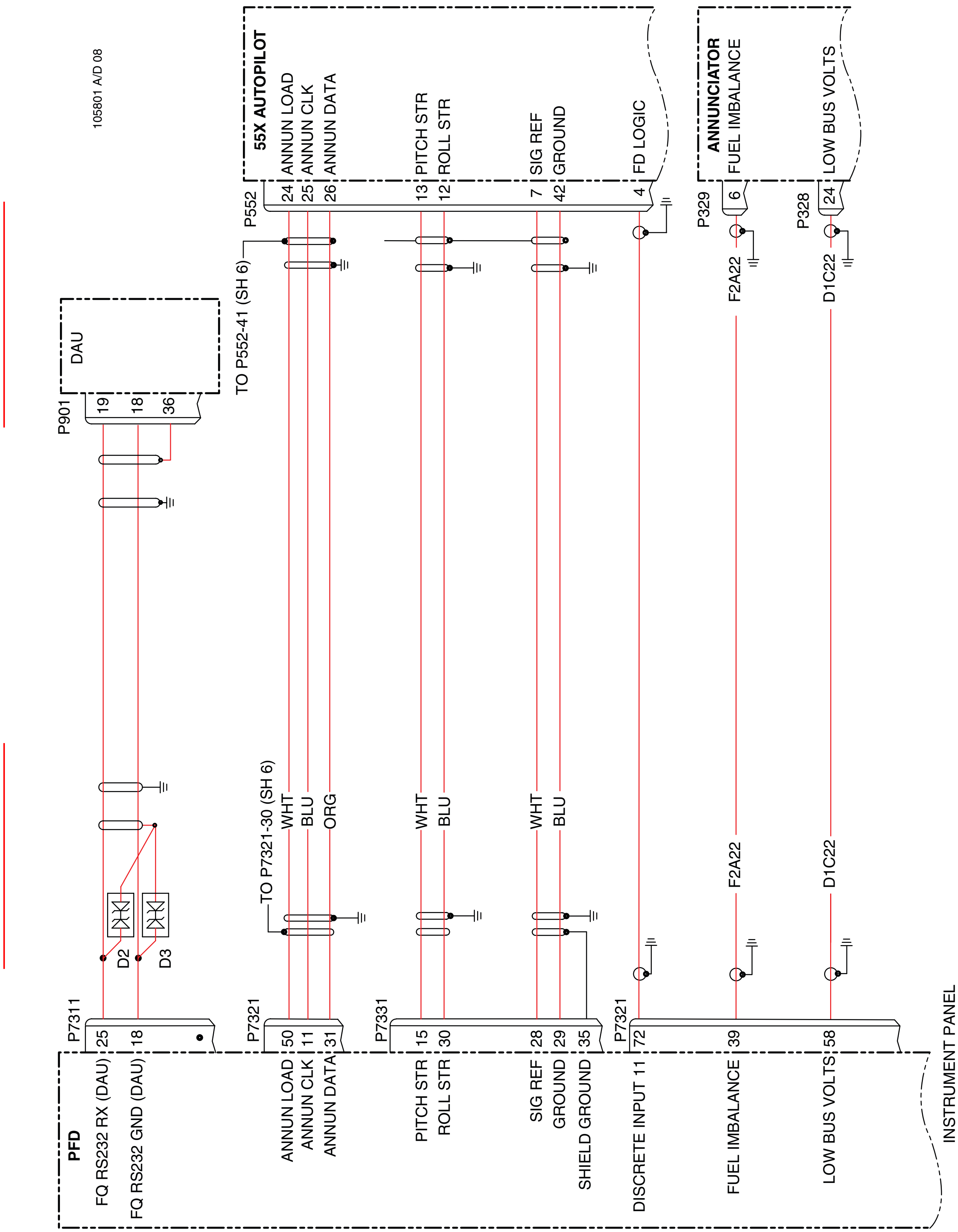
EFIS - Avidyne Entegra - Single PFD
Figure 6 (Sheet 1 of 4)



EFIS - Avidyne Entegra - Single PFD
Figure 6 (Sheet 2 of 4)



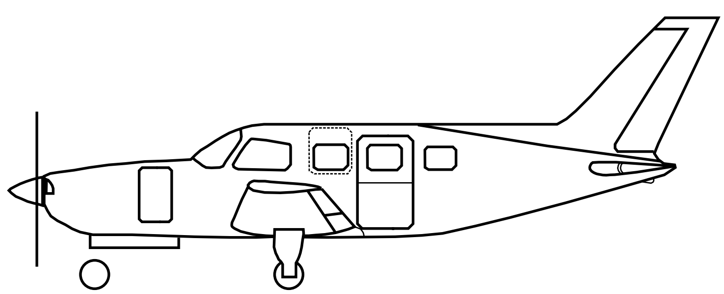
EFIS - Avidyne Entegra - Single PFD
Figure 6 (Sheet 3 of 4)



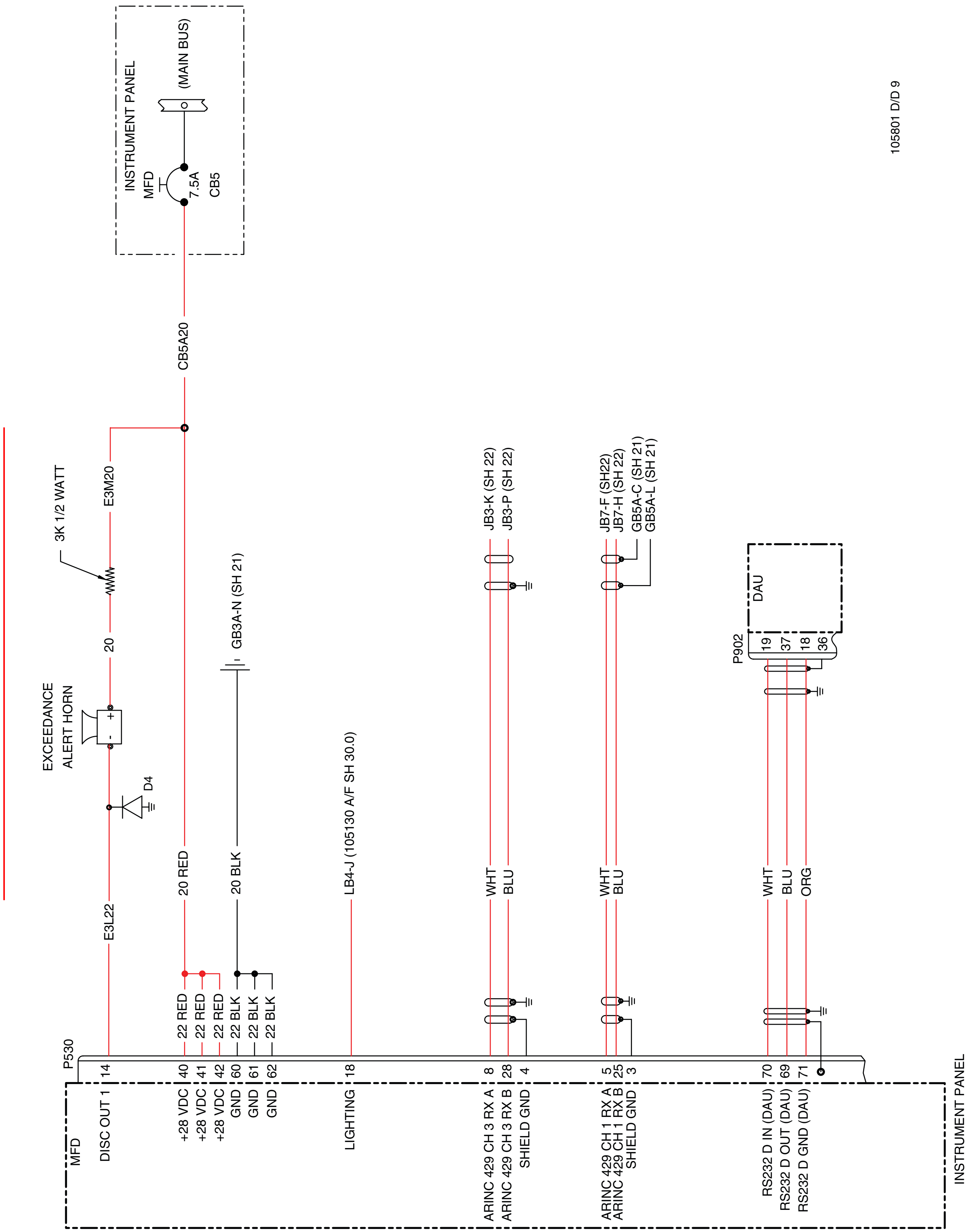
EFIS - Avidyne Entegra - Single PFD
Figure 6 (Sheet 4 of 4)

Item #	Designation	Description

Information Pending - See Parts Catalog



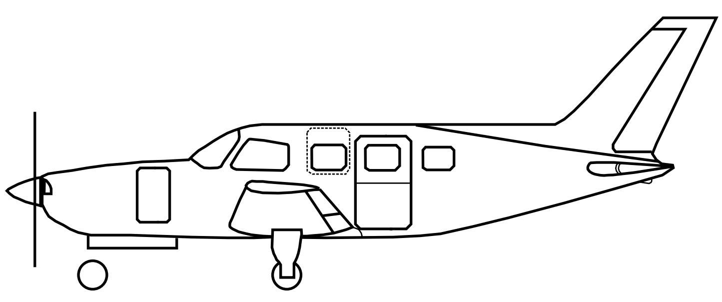
EFIS - Avidyne Entegra - MFD with Single PFD
Figure 7 (Sheet 1 of 2)



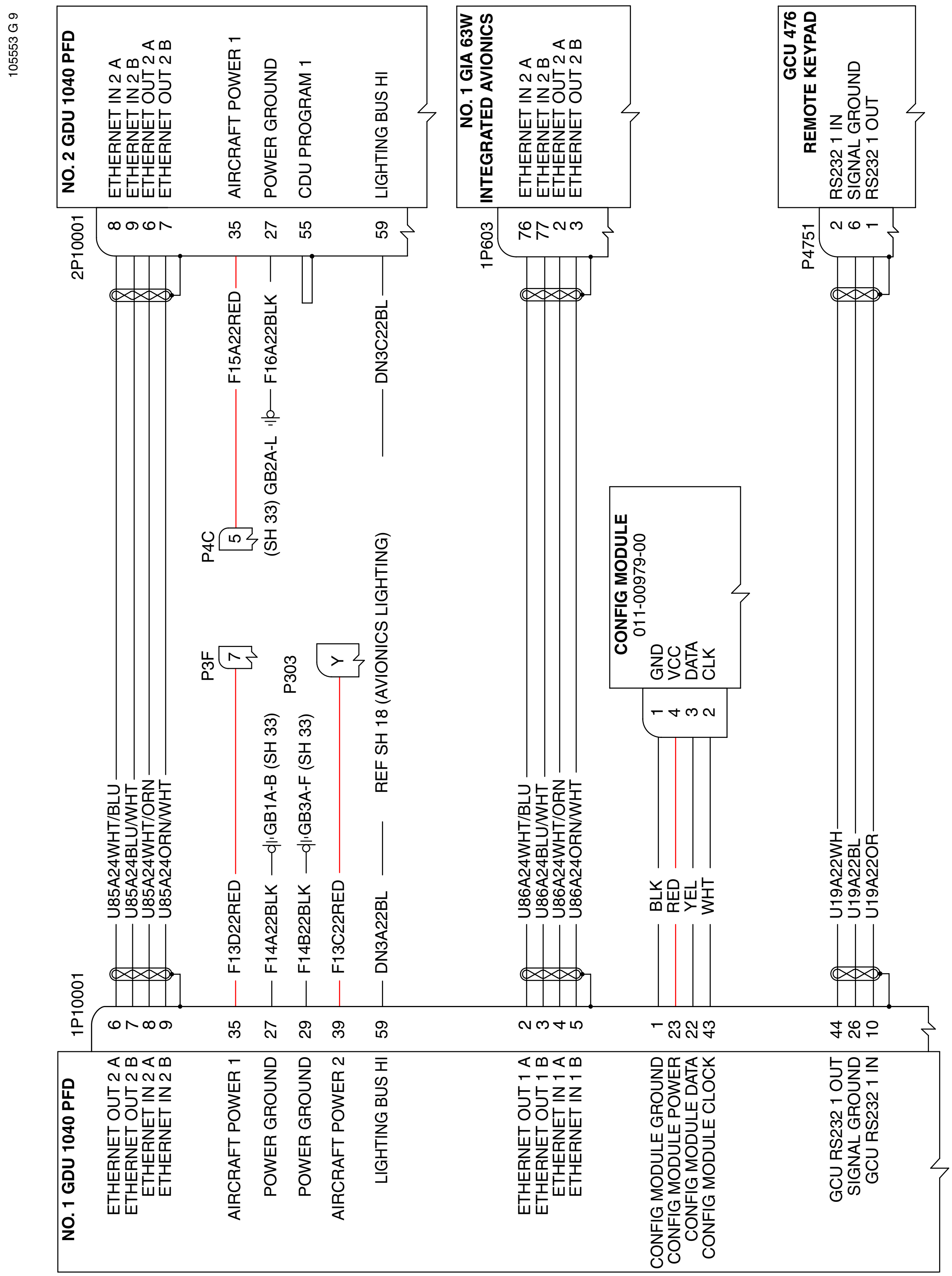
EFIS - Avidyne Entegra - MFD with Single PFD
Figure 7 (Sheet 2 of 2)

Item #	Designation	Description

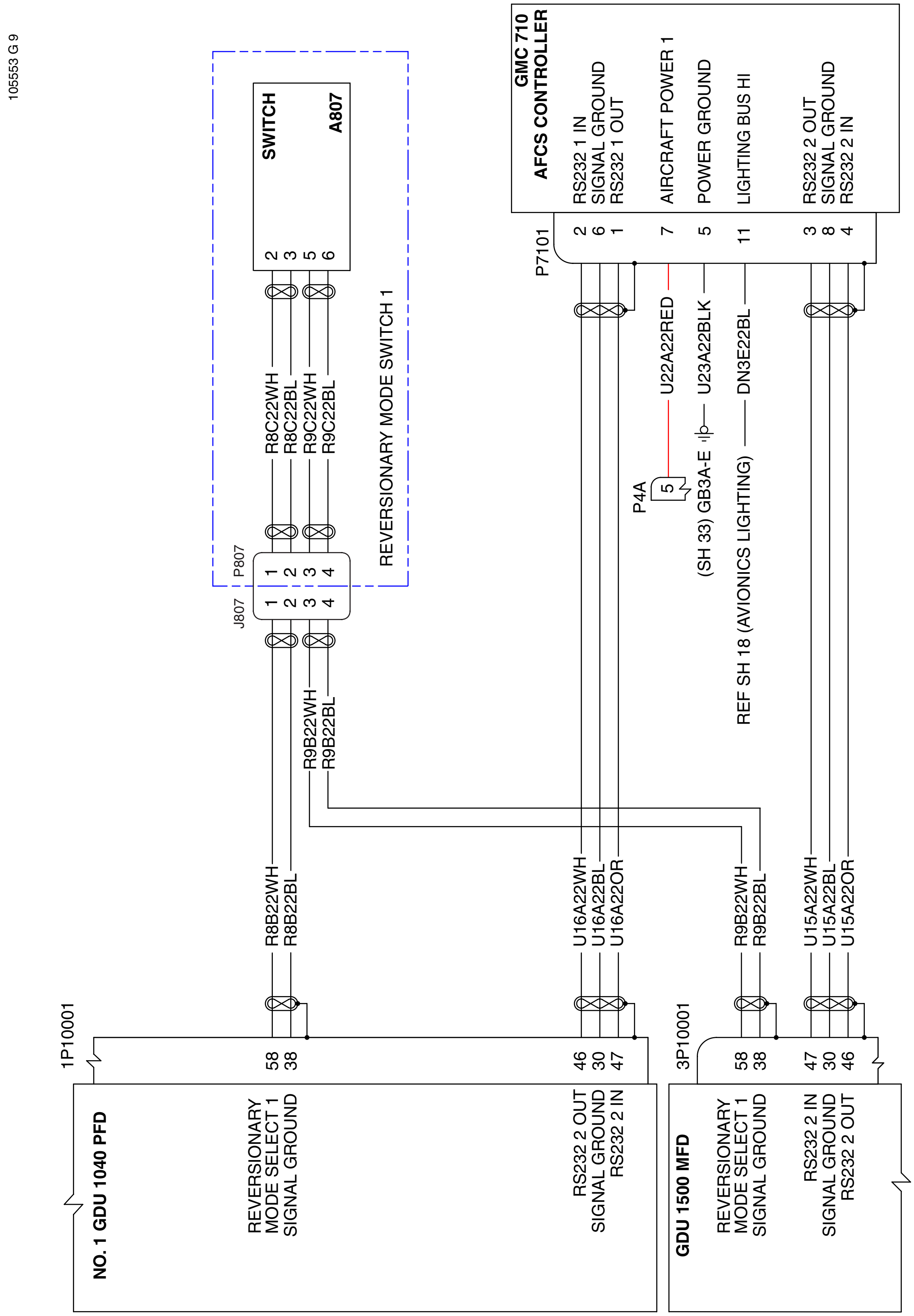
Information Pending - See Parts Catalog



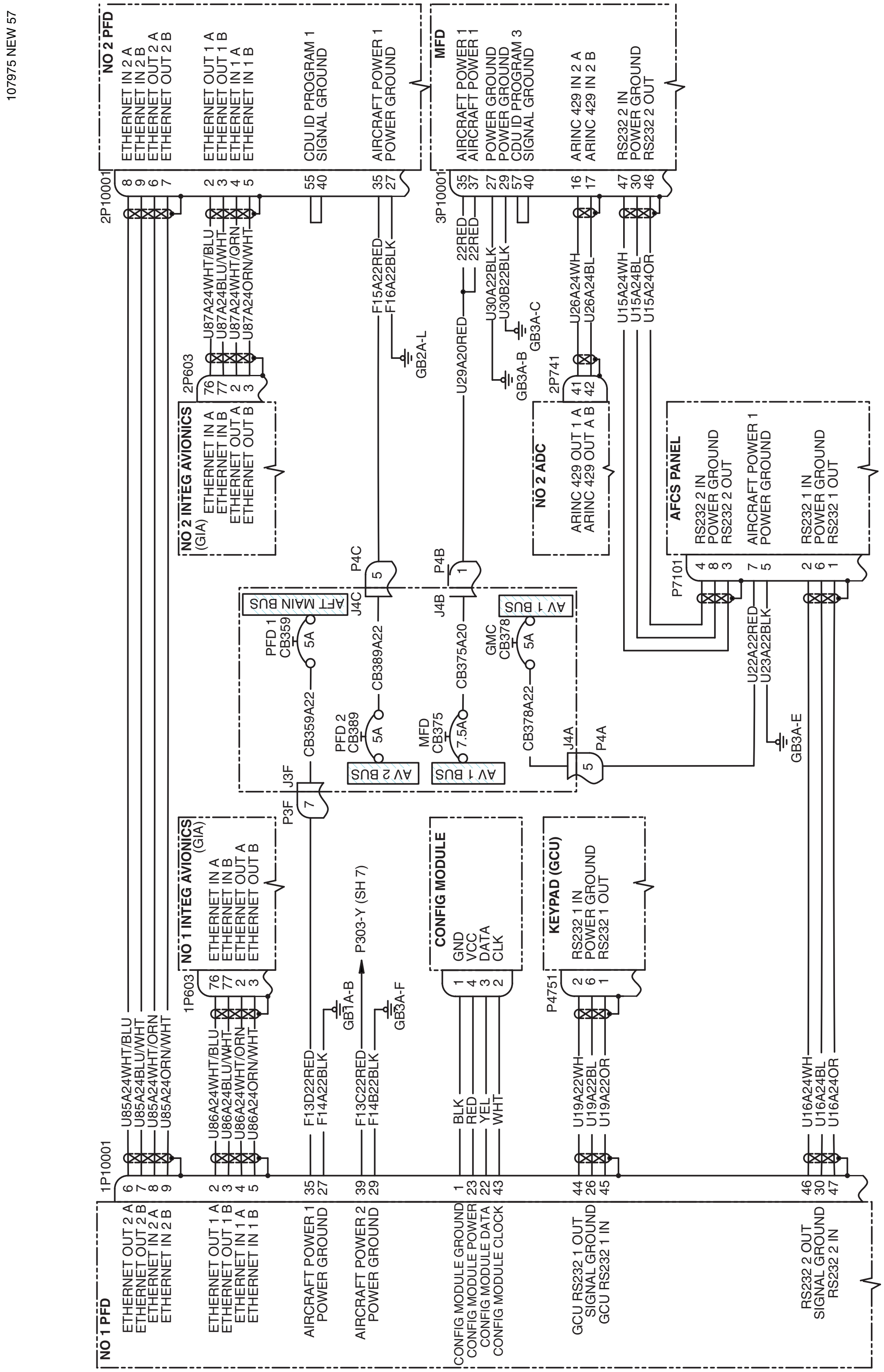
IAS - G1000 - PFD (GDU)
Figure 8 (Sheet 1 of 5)



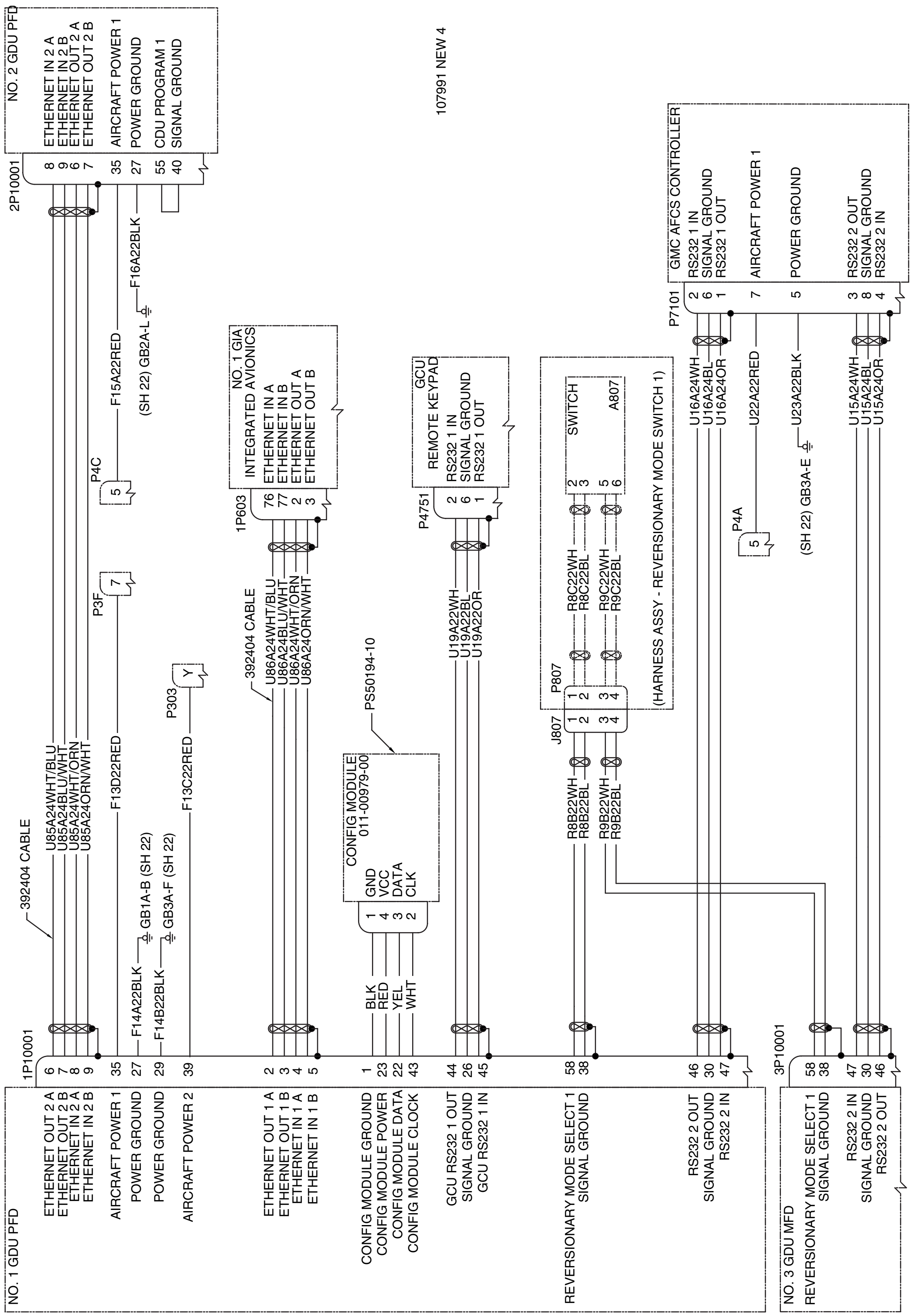
IAS - G1000 - PFD (GDU)
Figure 8 (Sheet 2 of 5)



IAS - G1000 - PFD (GDU)
Figure 8 (Sheet 3 of 5)



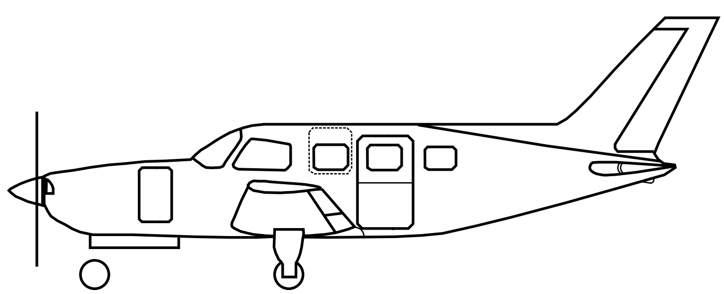
IAS - G1000 - PFD (GDU)
Figure 8 (Sheet 4 of 5)



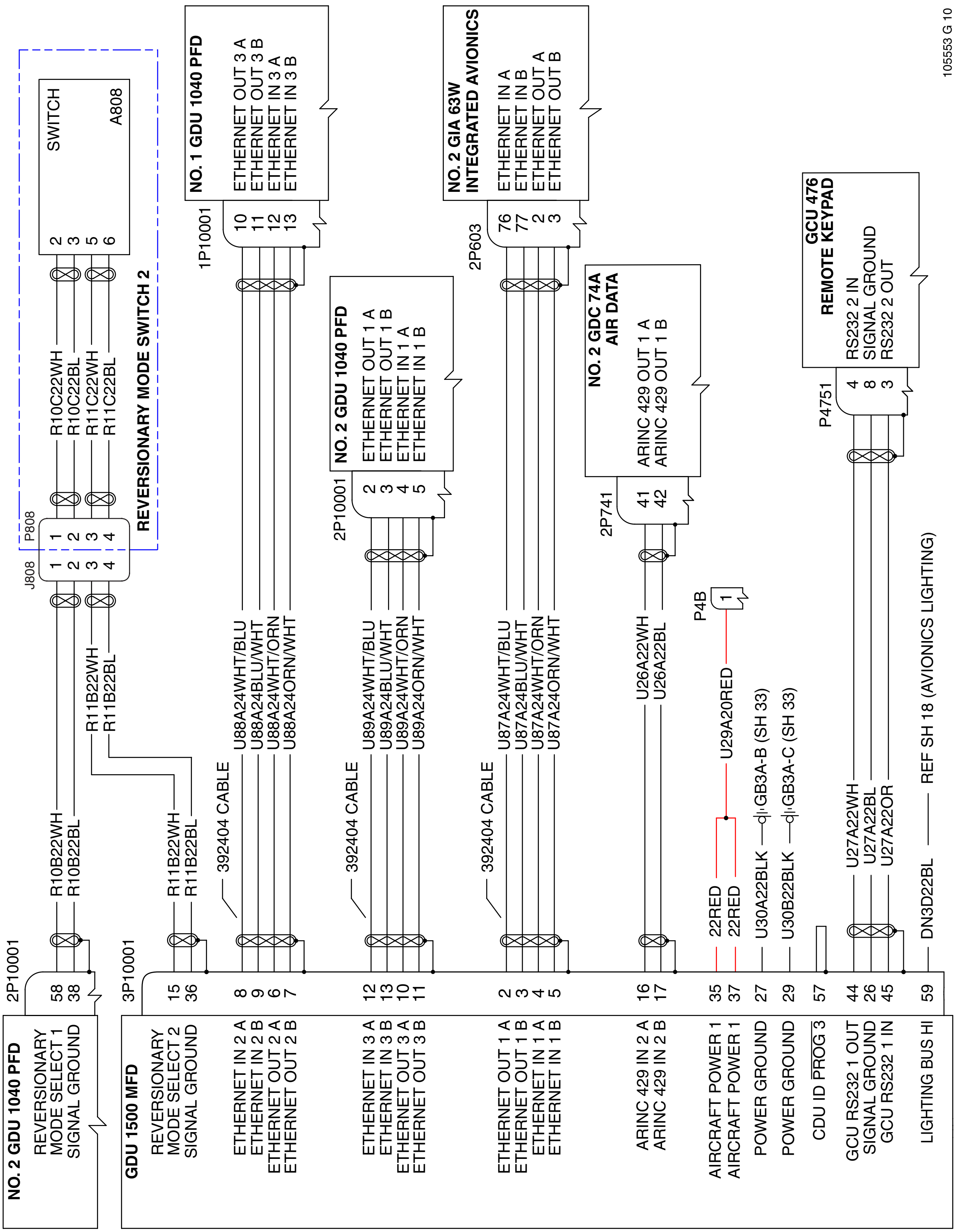
IAS - G1000 - PFD (GDU)
Figure 8 (Sheet 5 of 5)

Item #	Designation	Description

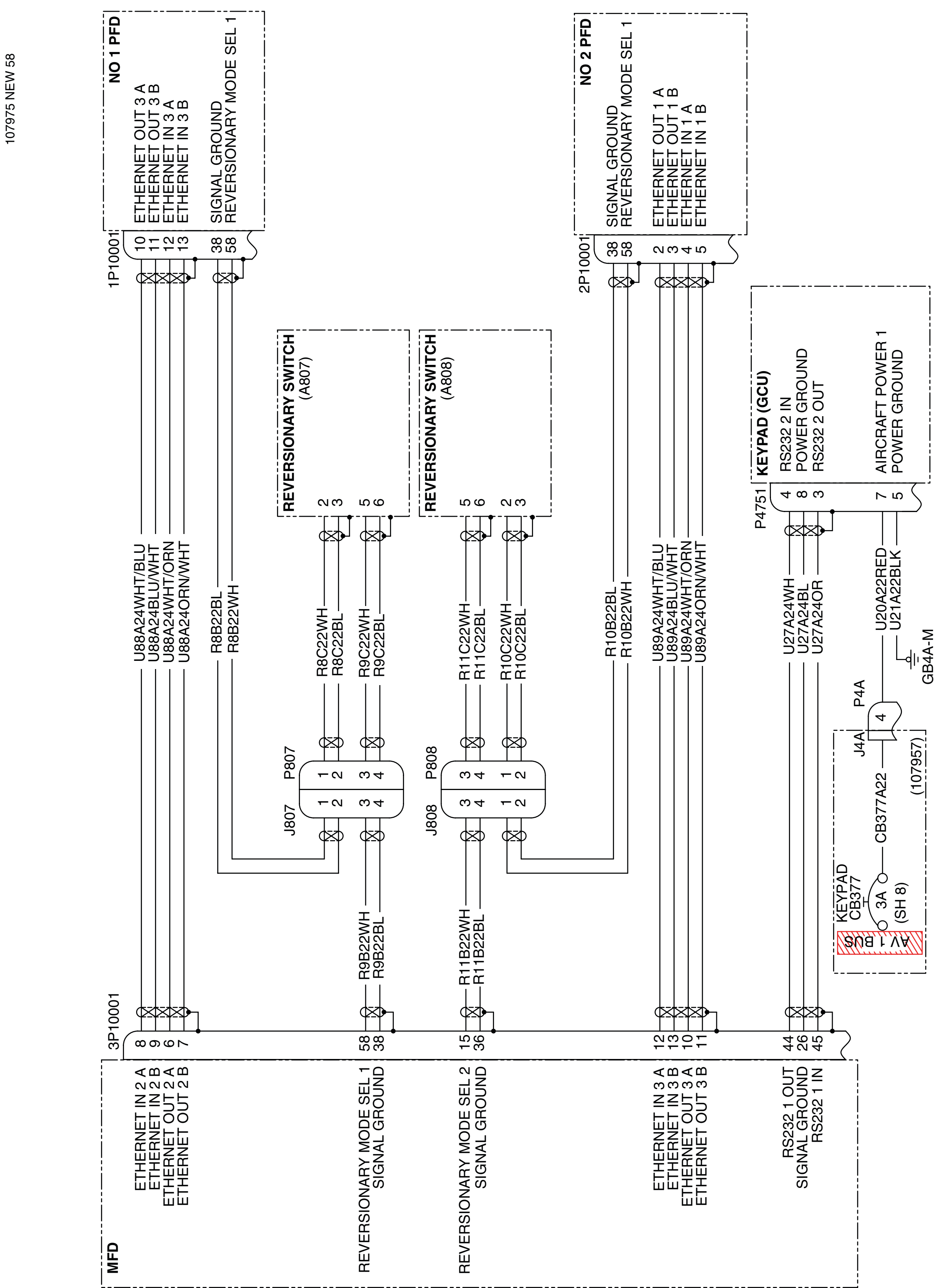
Information Pending - See Parts Catalog



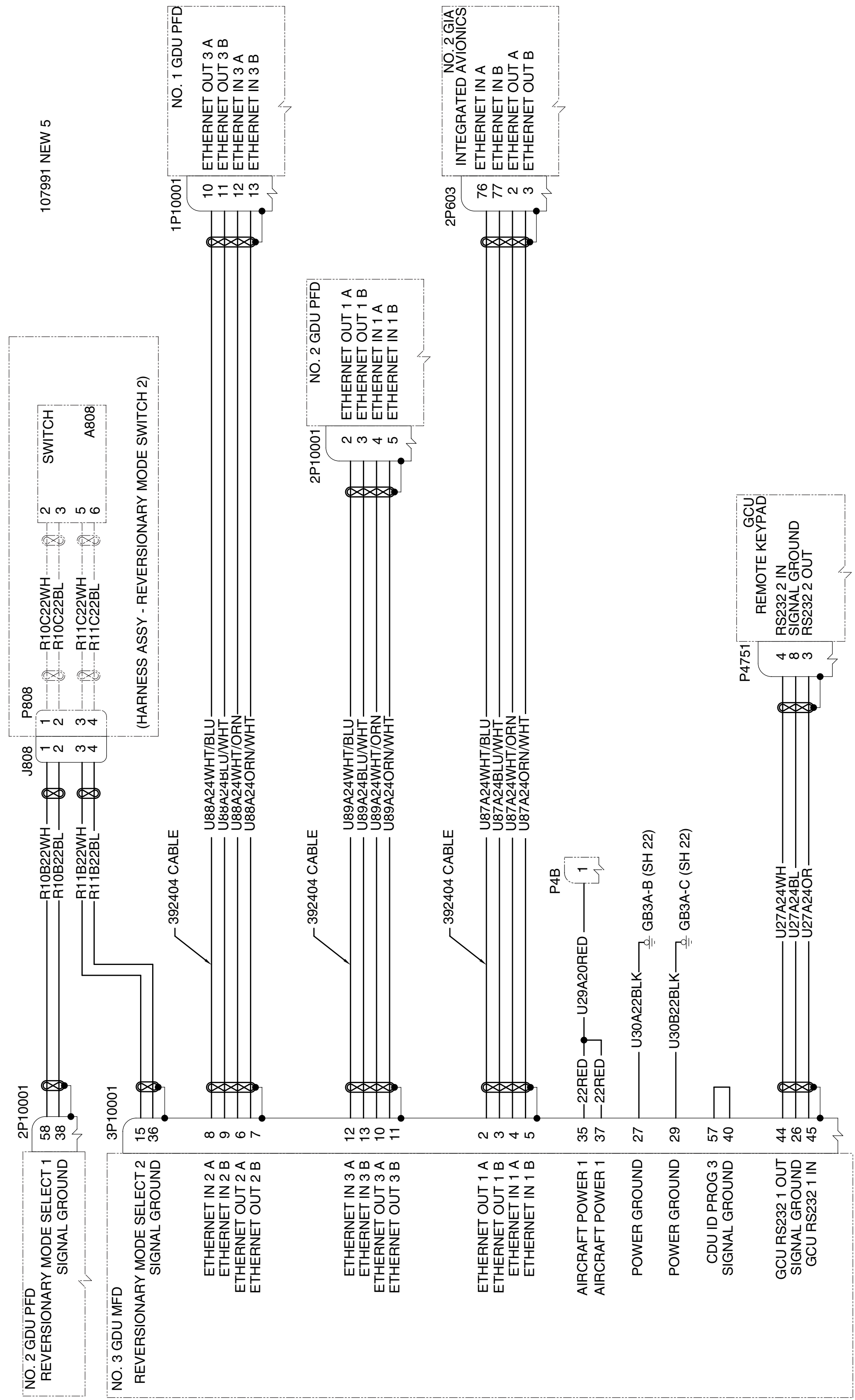
IAS - G1000 - MFD (GDU)
Figure 9 (Sheet 1 of 4)



IAS - G1000 - MFD (GDU)
Figure 9 (Sheet 2 of 4)



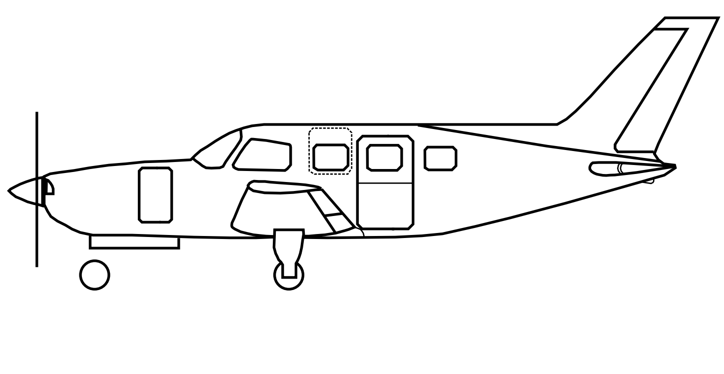
IAS - G1000 - MFD (GDU)
Figure 9 (Sheet 3 of 4)



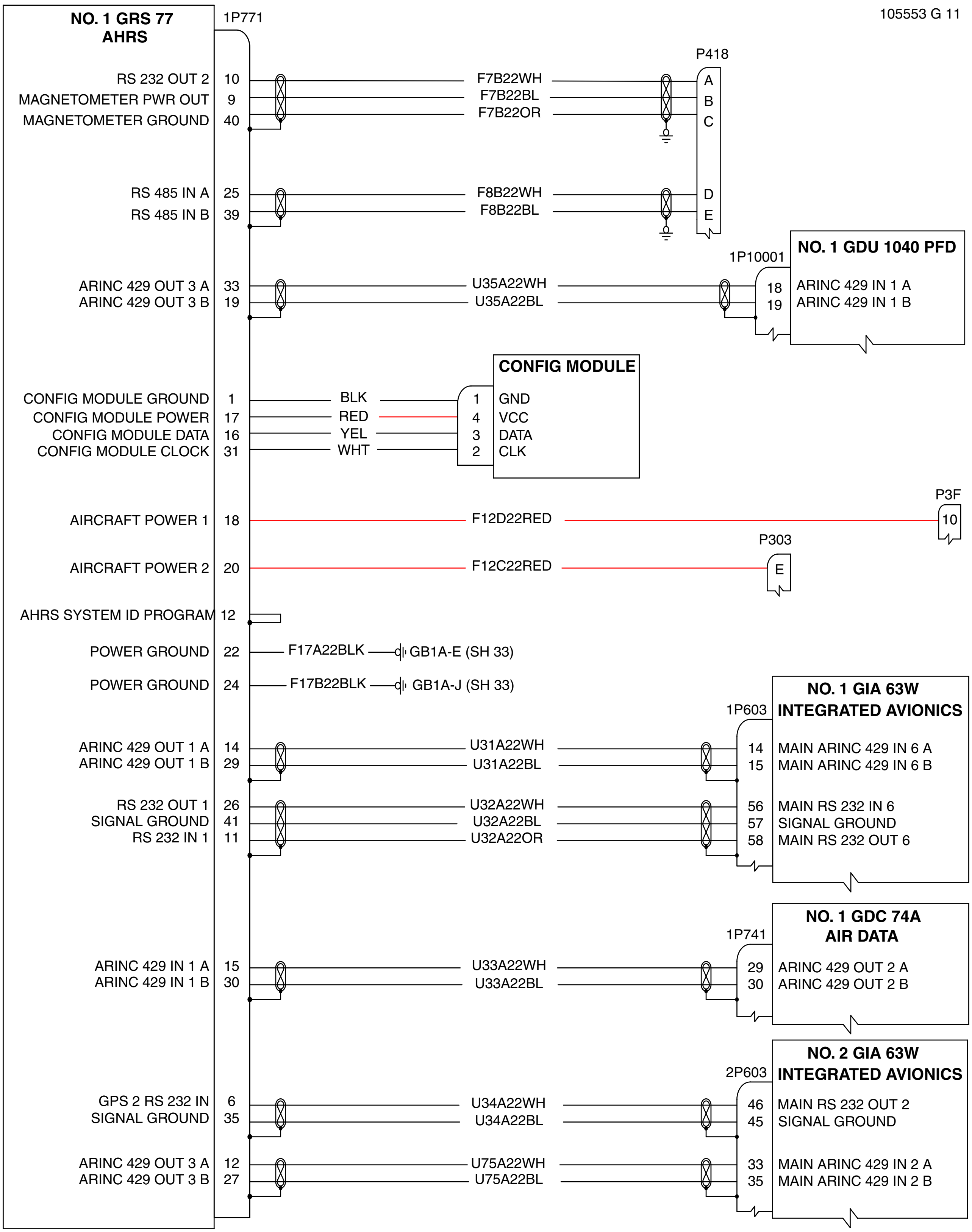
IAS - G1000 - MFD (GDU)
Figure 9 (Sheet 4 of 4)

Item #	Designation	Description

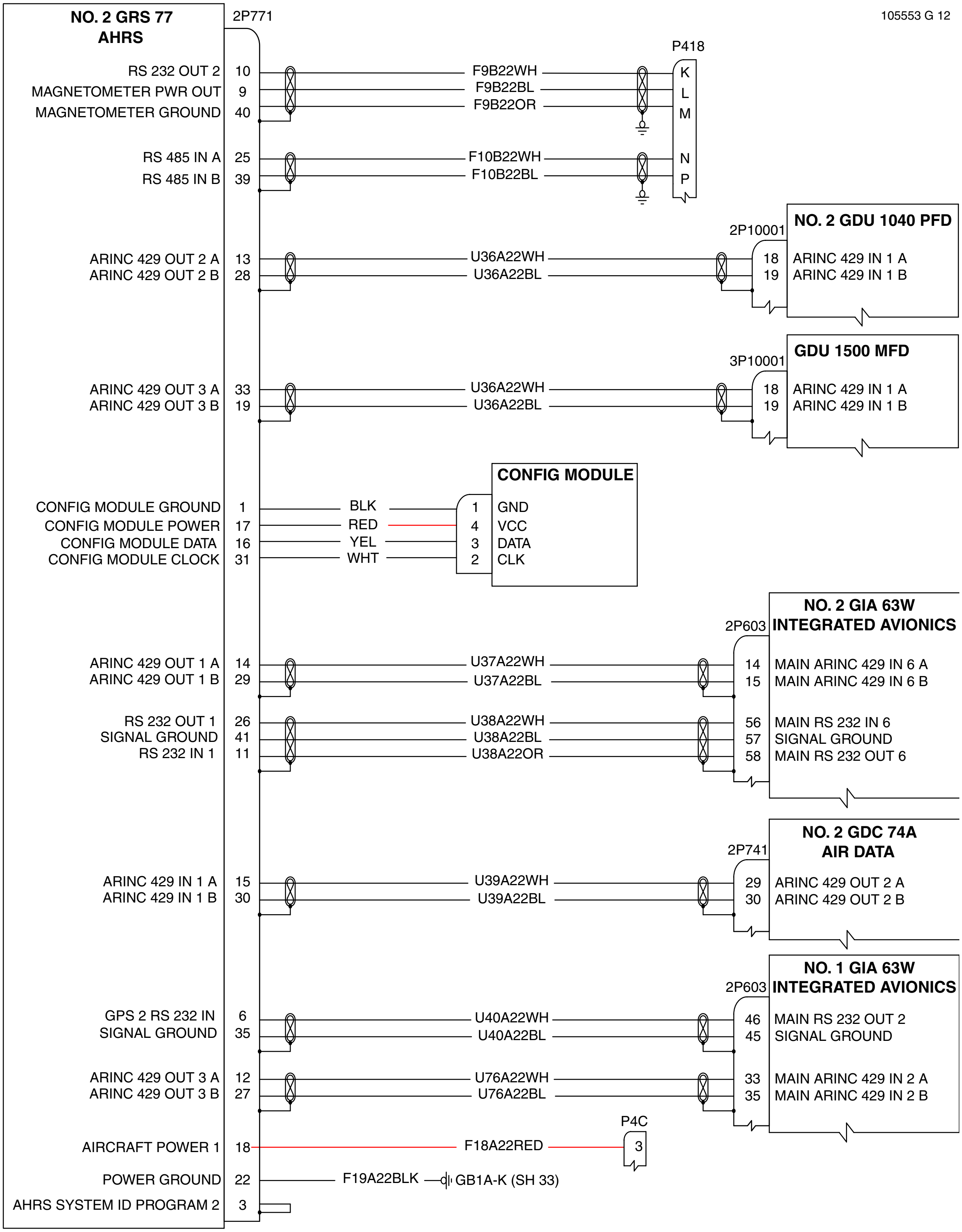
Information Pending - See Parts Catalog



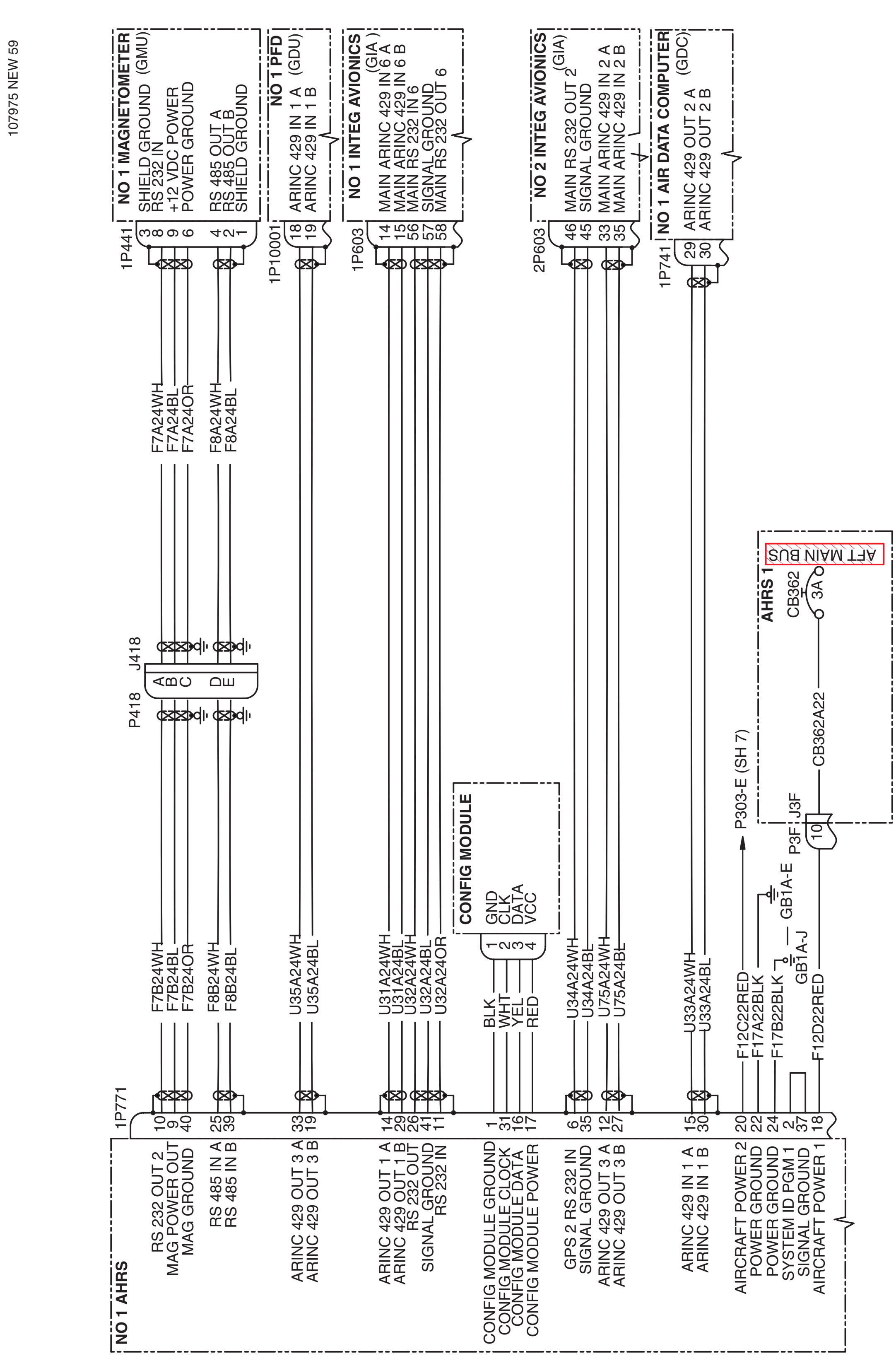
IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 1 of 8)



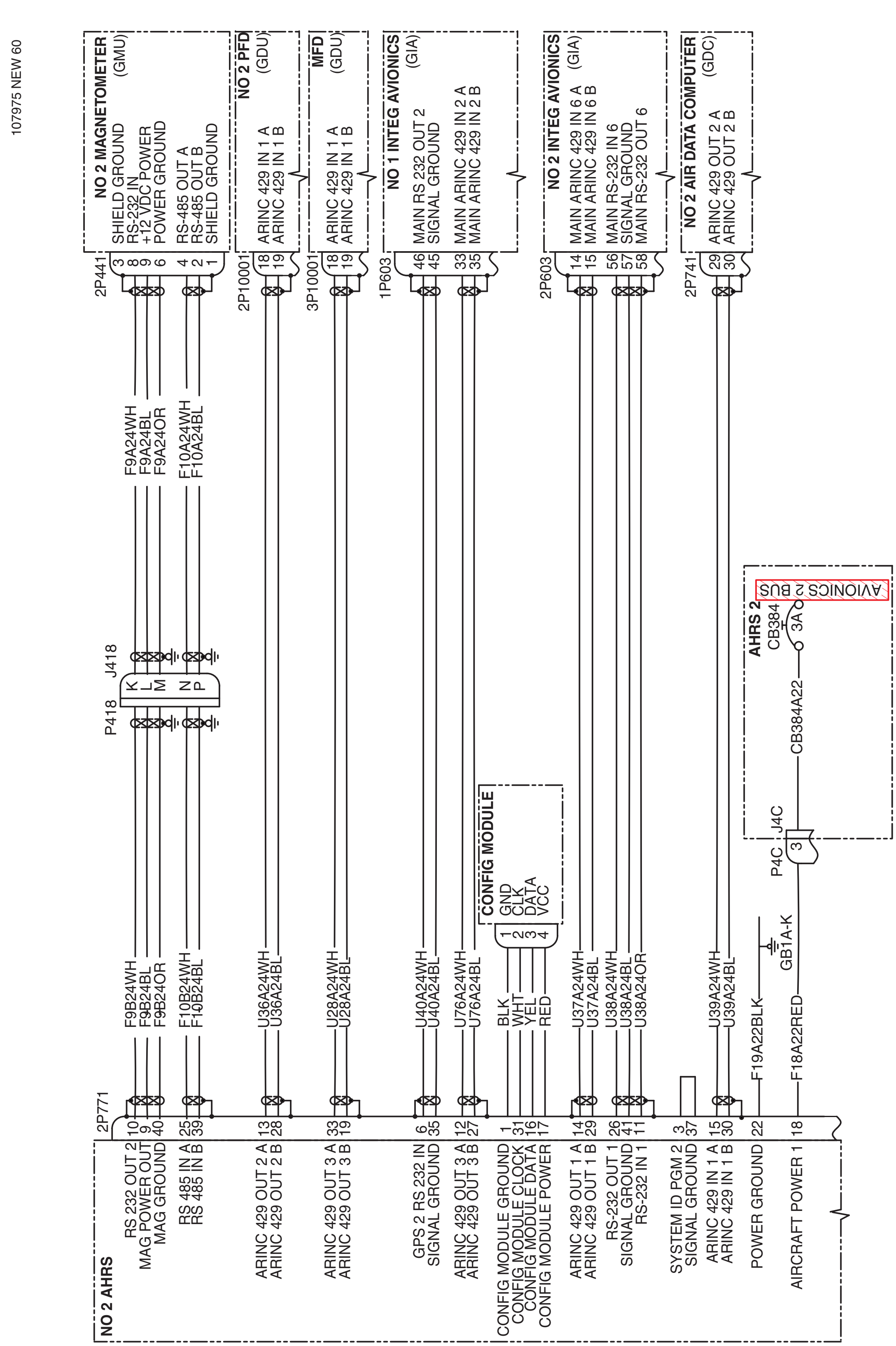
IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 2 of 8)



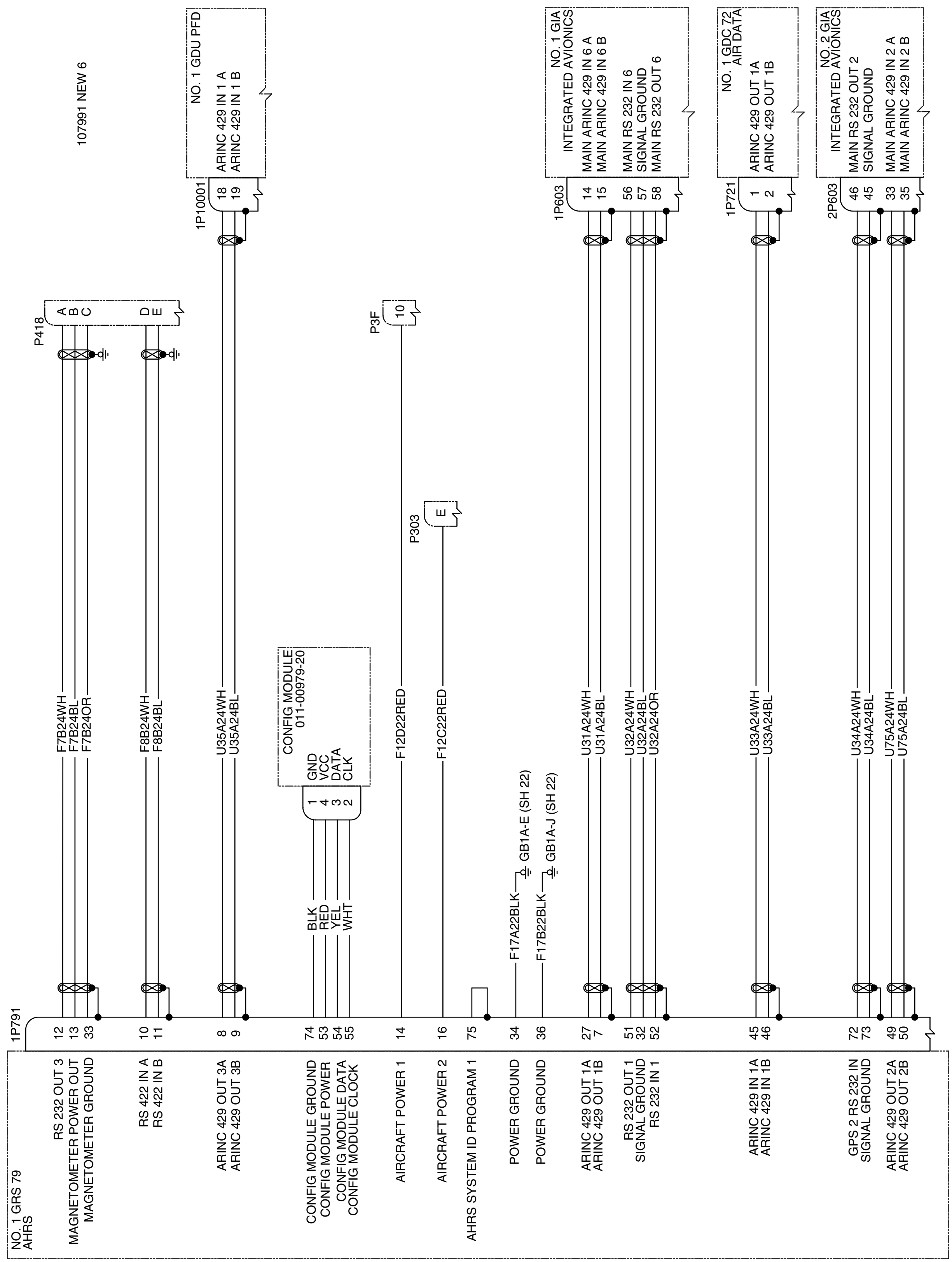
IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 3 of 8)



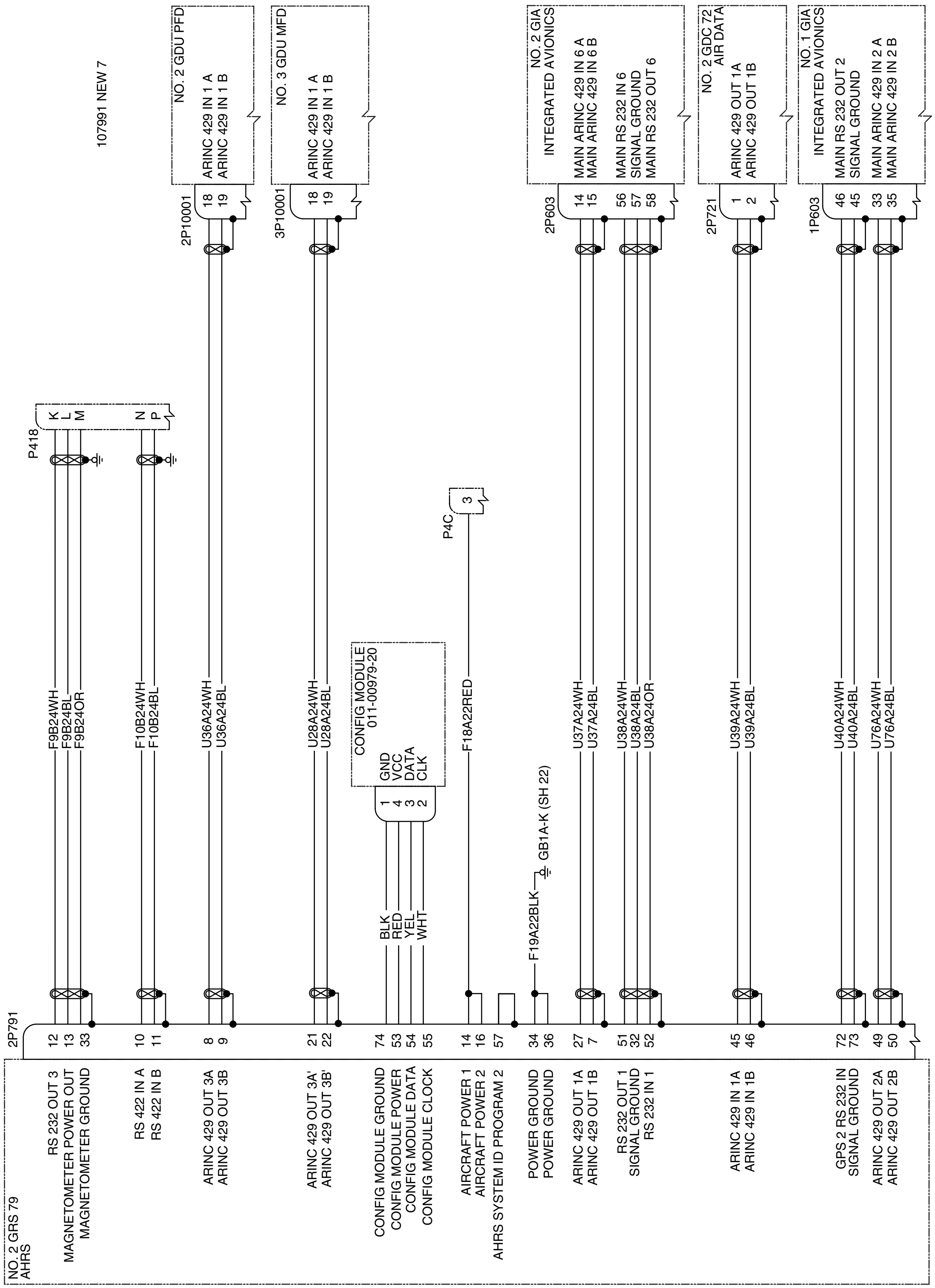
IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 4 of 8)



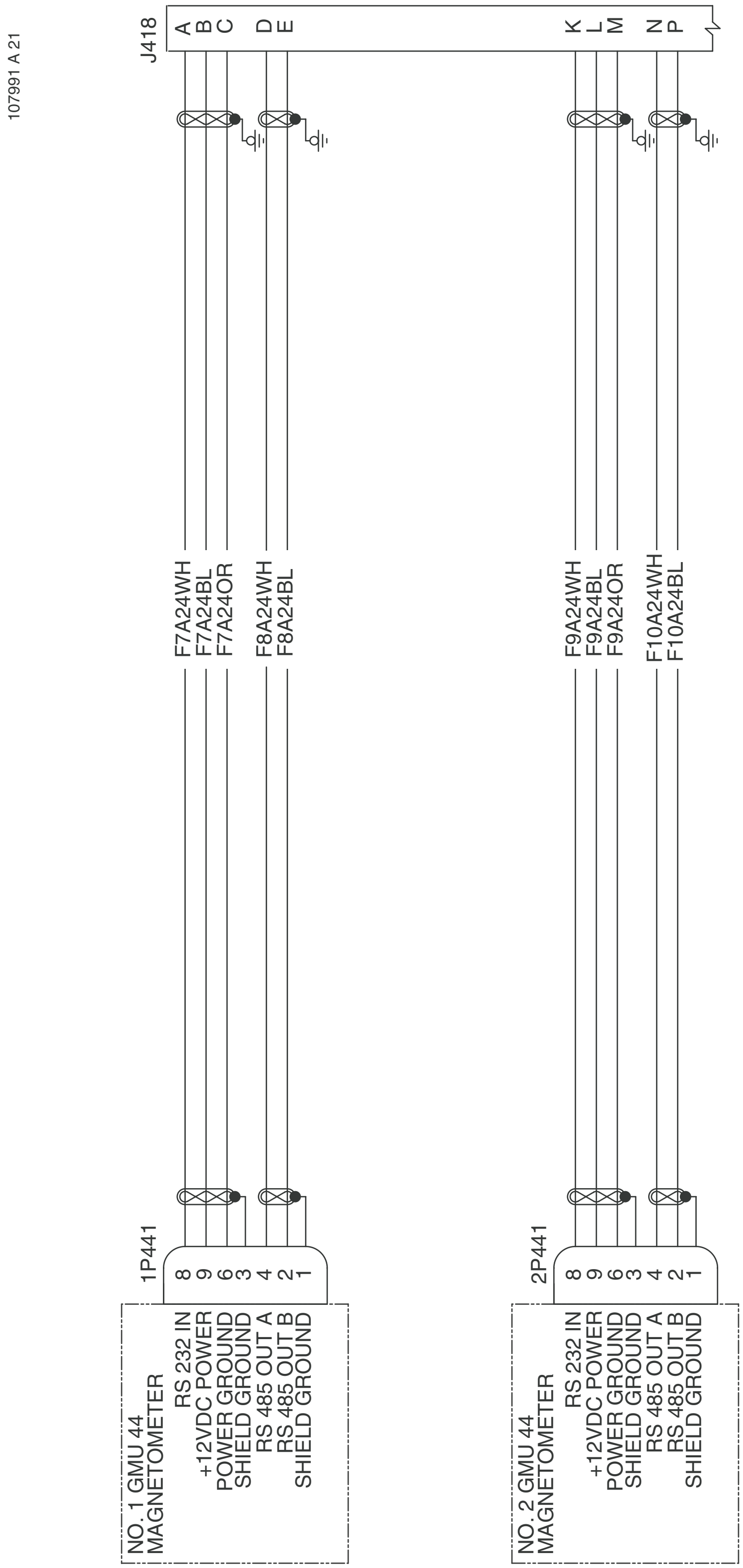
IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 5 of 8)



IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 6 of 8)



IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 7 of 8)



IAS - G1000 - AHRS (GRS)
Figure 10 (Sheet 8 of 8)

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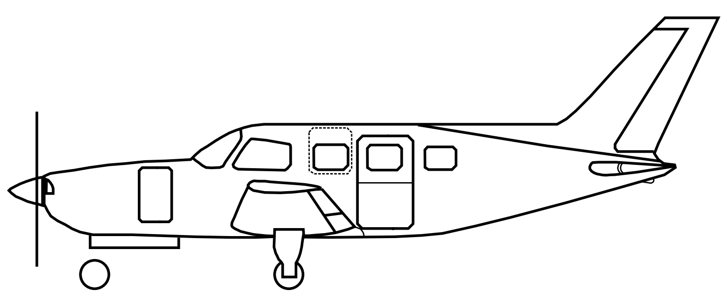
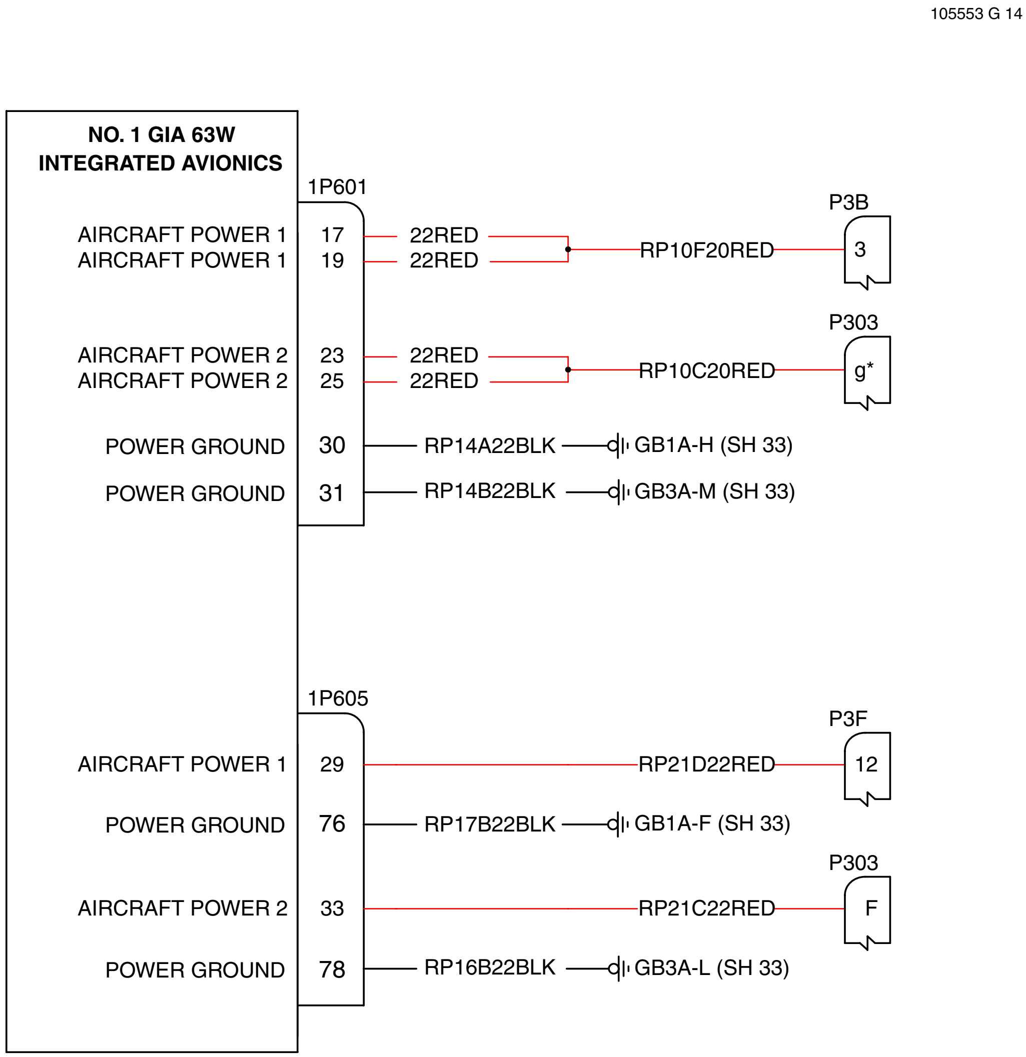
Item #	Designation	Description

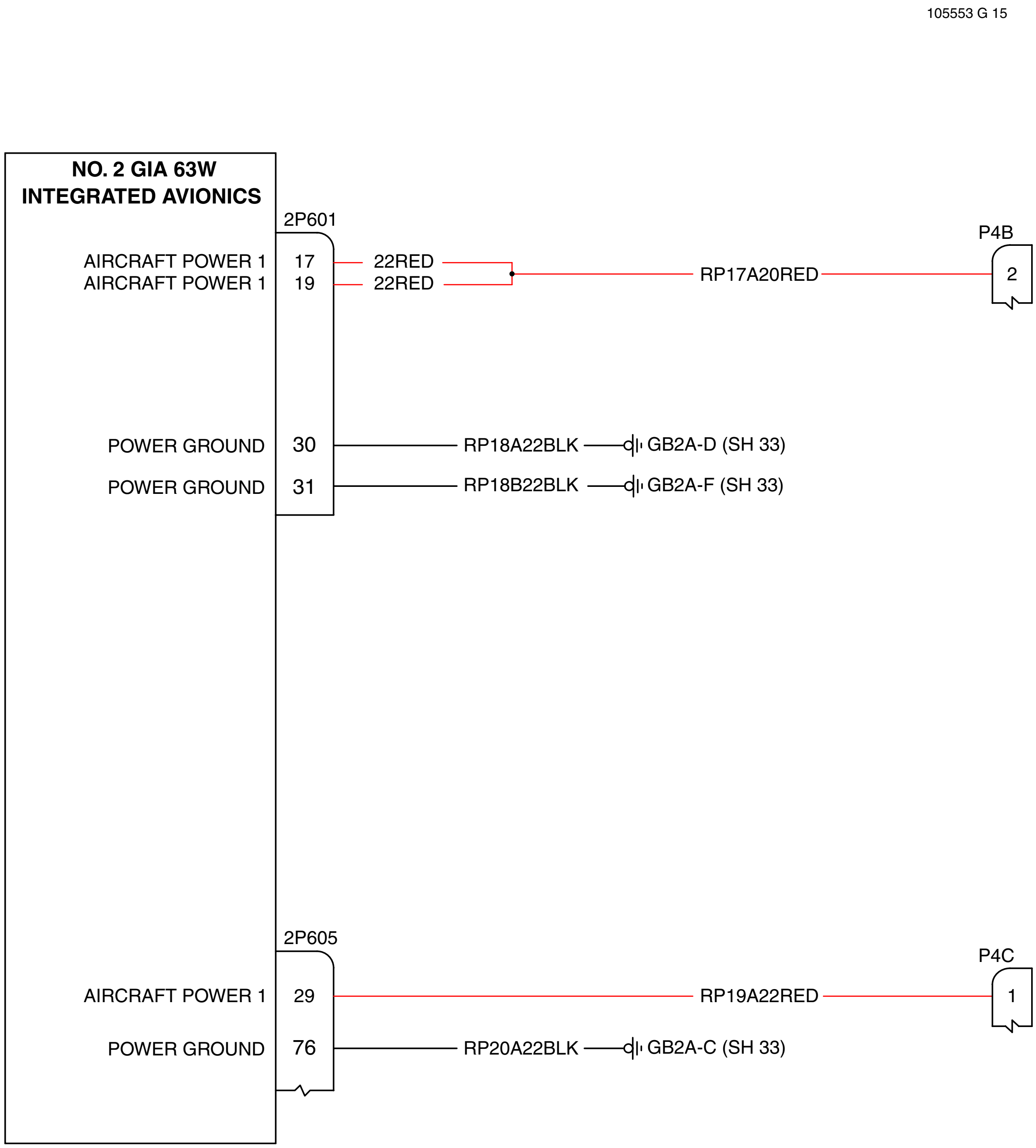
Information Pending - See Parts Catalog



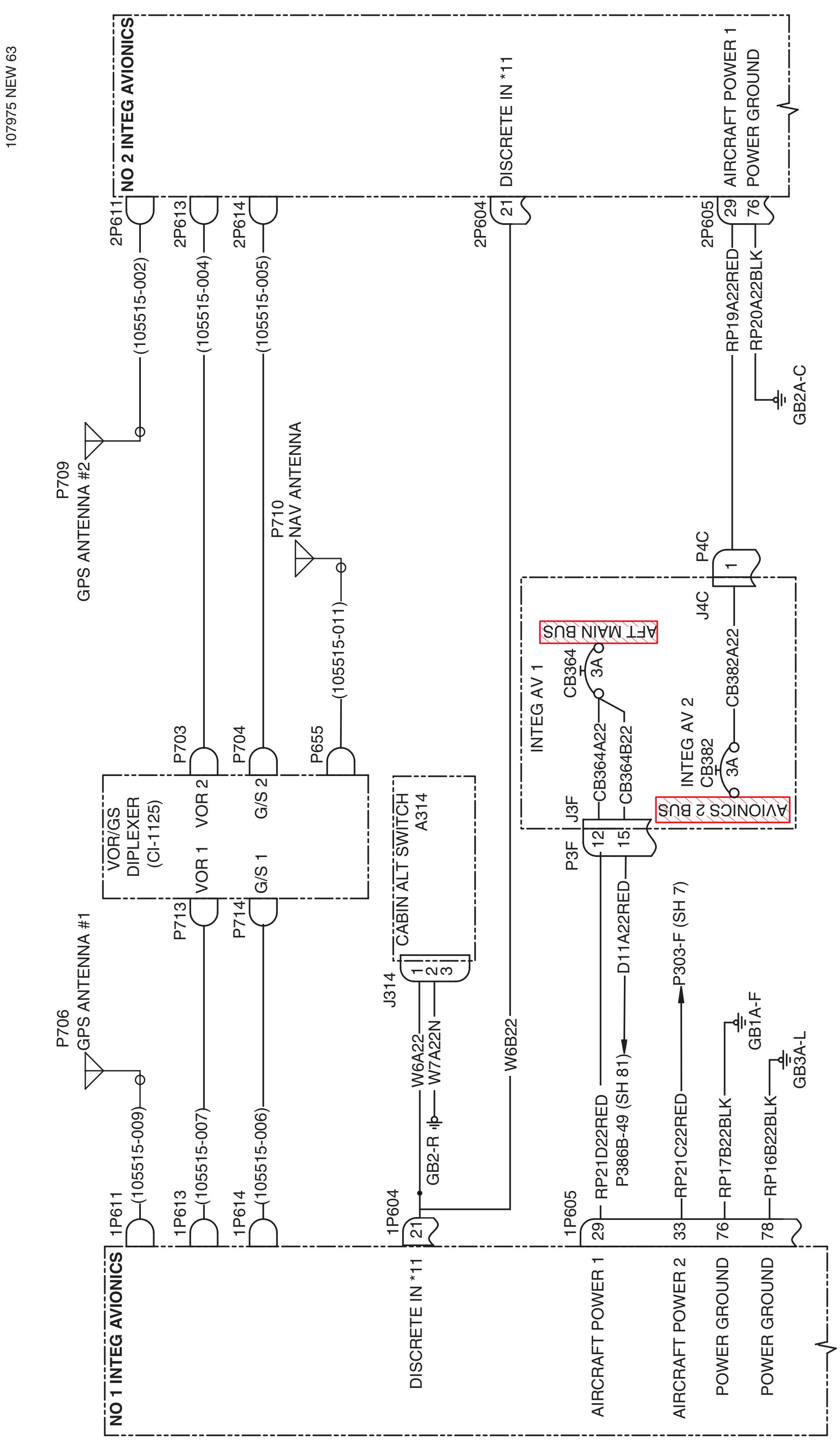
IAS - G1000 - Integrated Avionics (GIA)
Figure 11 (Sheet 1 of 7)



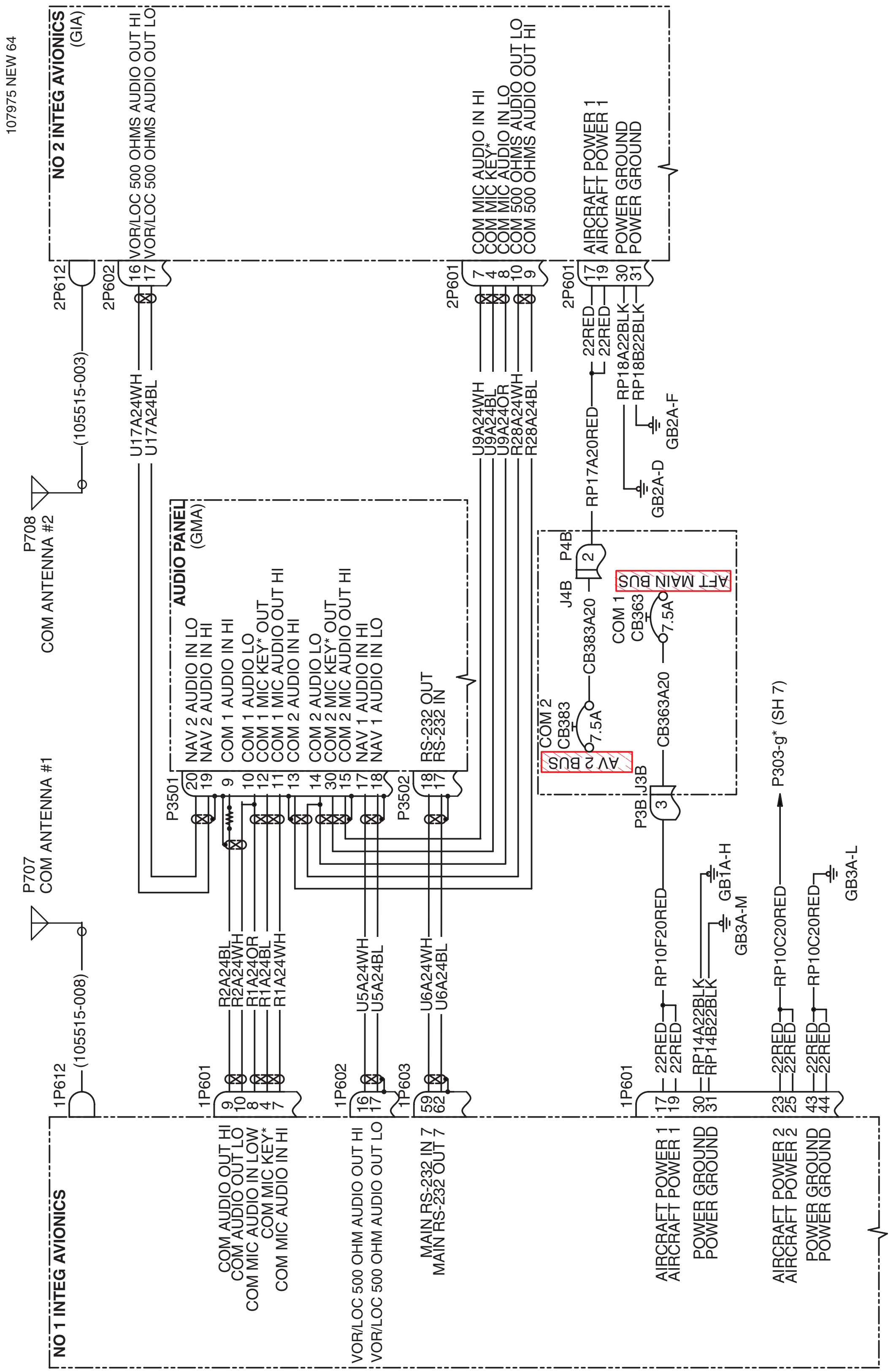
IAS - G1000 - Integrated Avionics (GIA)
Figure 11 (Sheet 2 of 7)



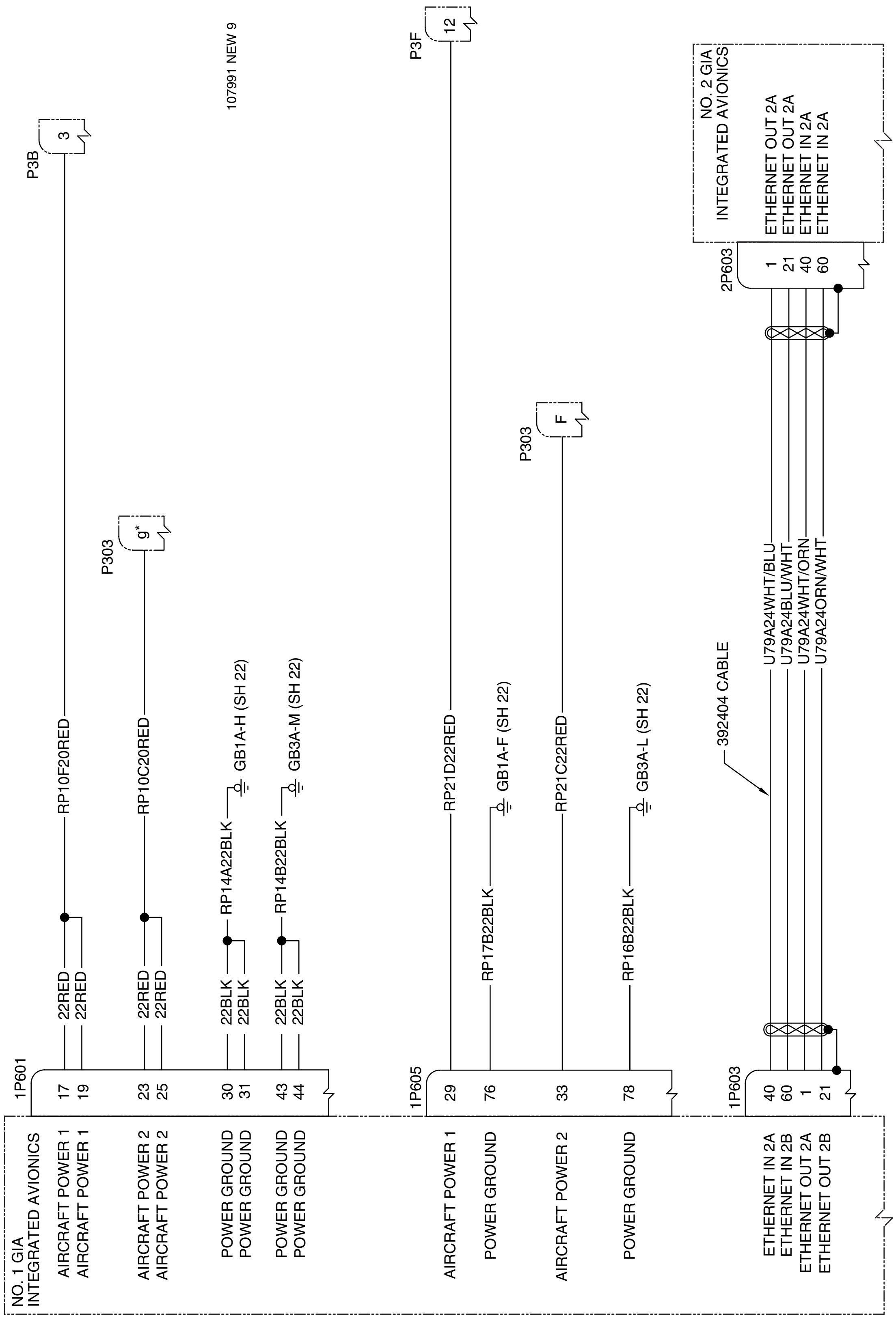
IAS - G1000 - Integrated Avionics (GIA)
Figure 11 (Sheet 3 of 7)



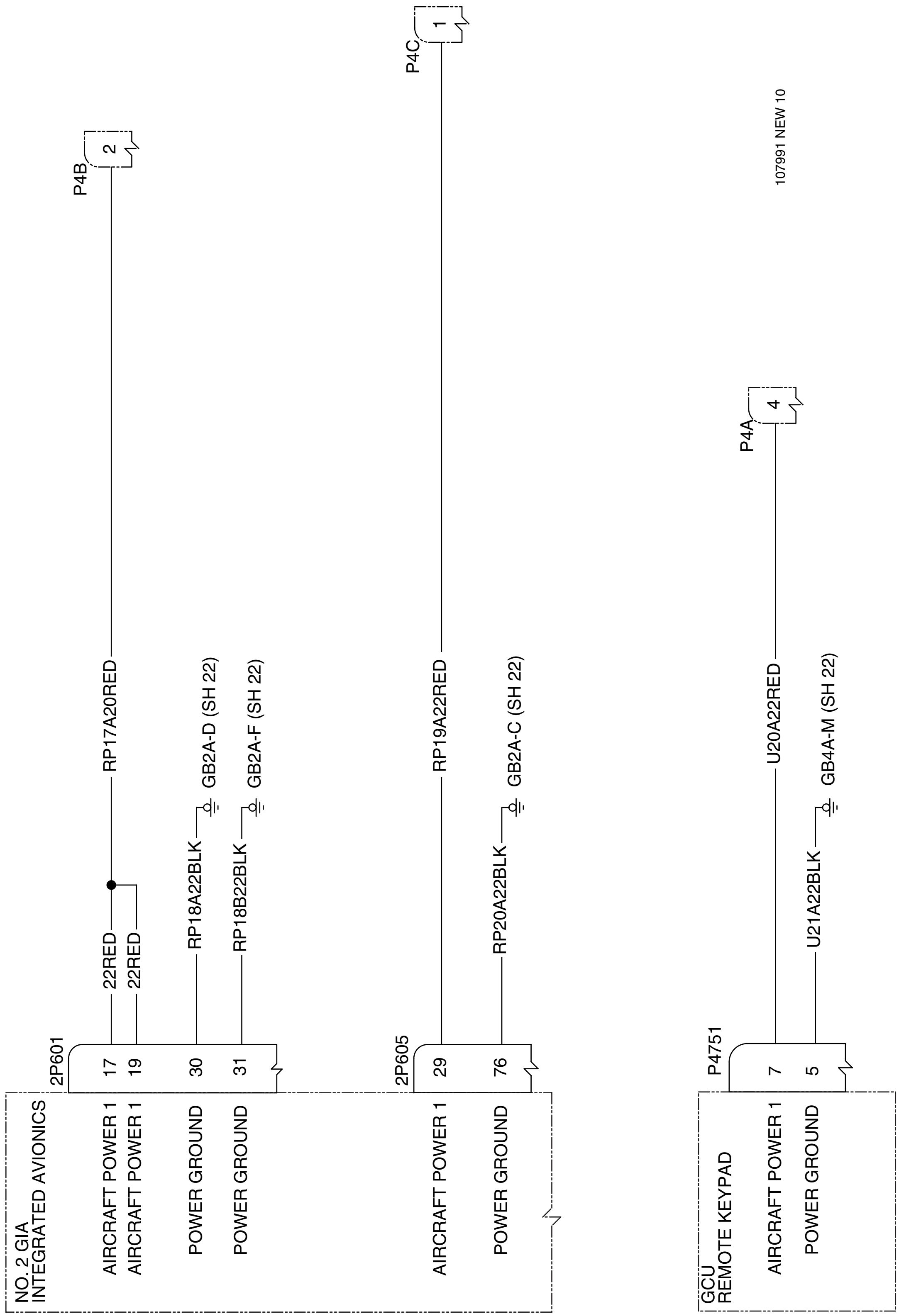
IAS - G1000 - Integrated Avionics (GIA)
Figure 11 (Sheet 4 of 7)



IAS - G1000 - Integrated Avionics (GIA)
Figure 11 (Sheet 5 of 7)



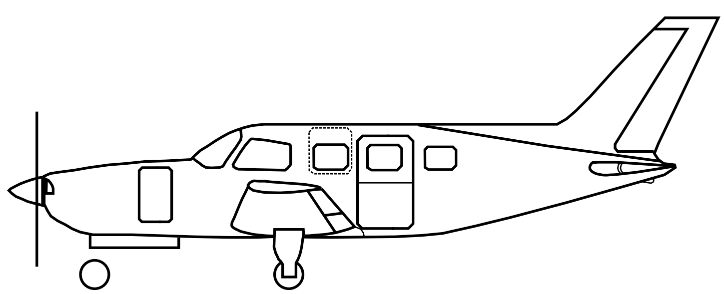
IAS - G1000 - Integrated Avionics (GIA)
Figure 11 (Sheet 6 of 7)



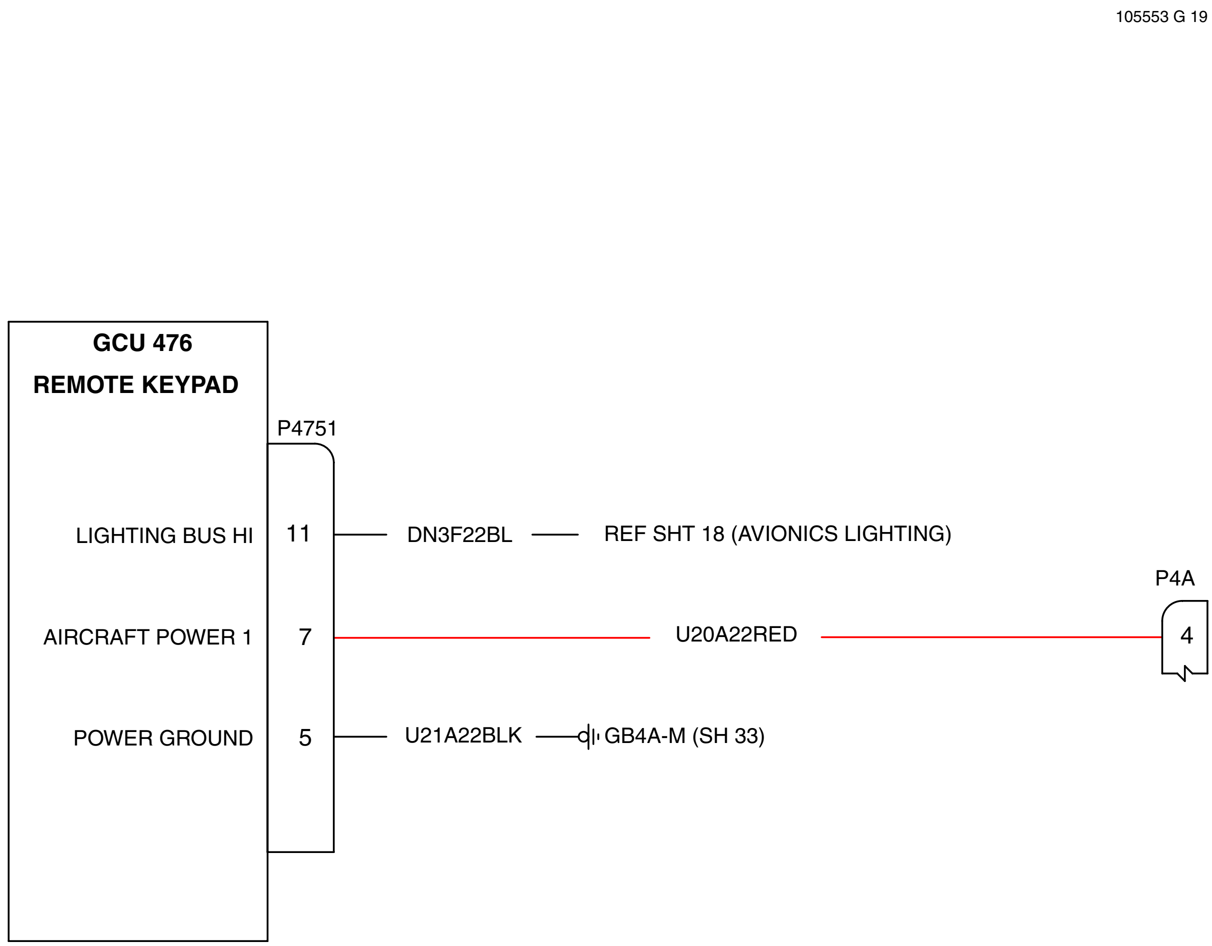
IAS - G1000 - Integrated Avionics (GIA)
Figure 11 (Sheet 7 of 7)

Item #	Designation	Description

Information Pending - See Parts Catalog



IAS - G1000 - Remote Keypad (GCU 476)
Figure 12 (Sheet 1 of 2)

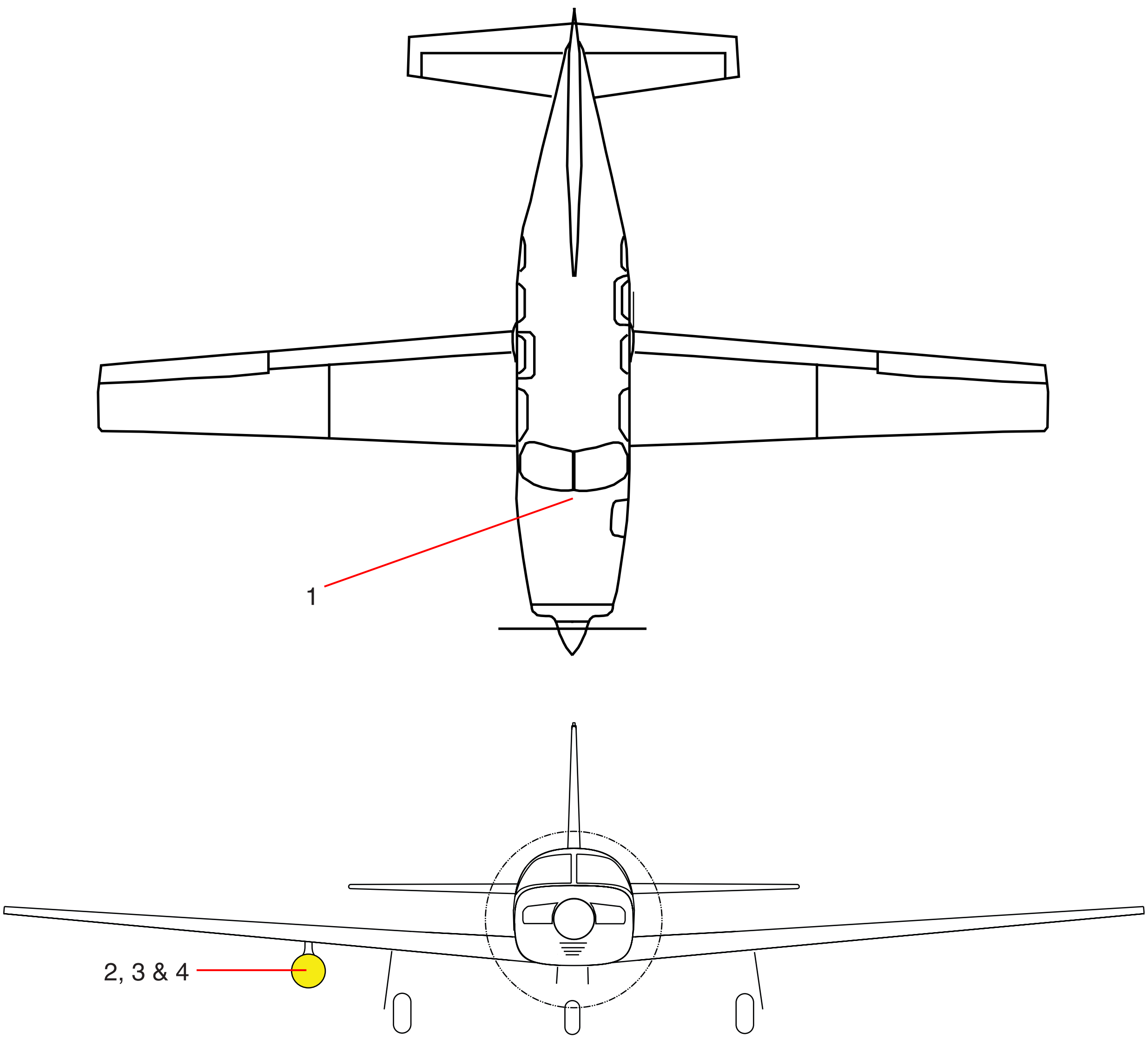


See 107975 SH 58 for S/N's 4636633, 4636652 and up.

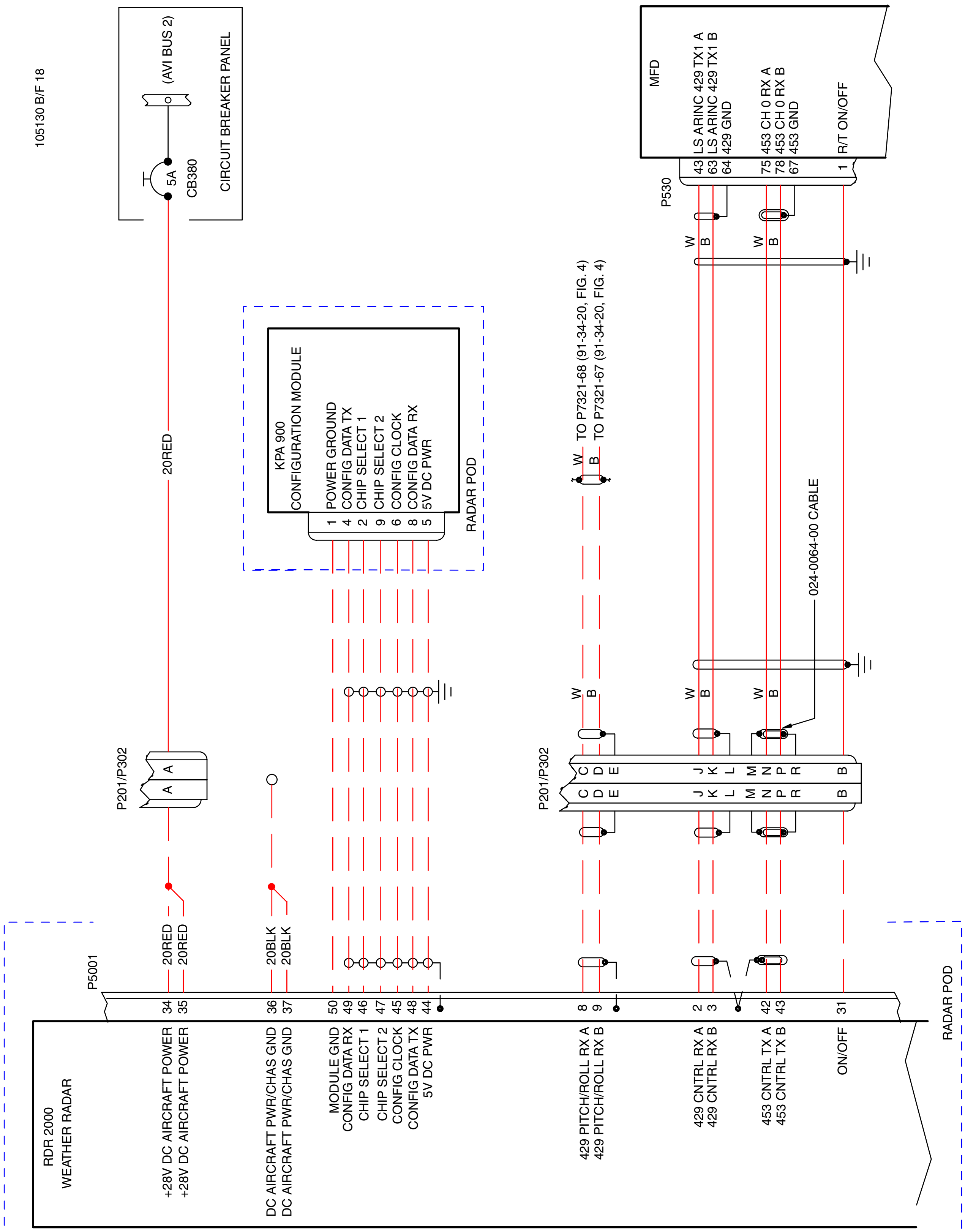
IAS - G1000 - Remote Keypad (GCU 476)
Figure 12 (Sheet 2 of 2)

Item #	Designation	Description
1	IN-182A or MFD	Radar Indicator
2	ART 2000 *	Radar Antenna Transceiver
3	AA201 OH or GWX-68	Radar Antenna Array
4	CM 2000 *	Configuration Module

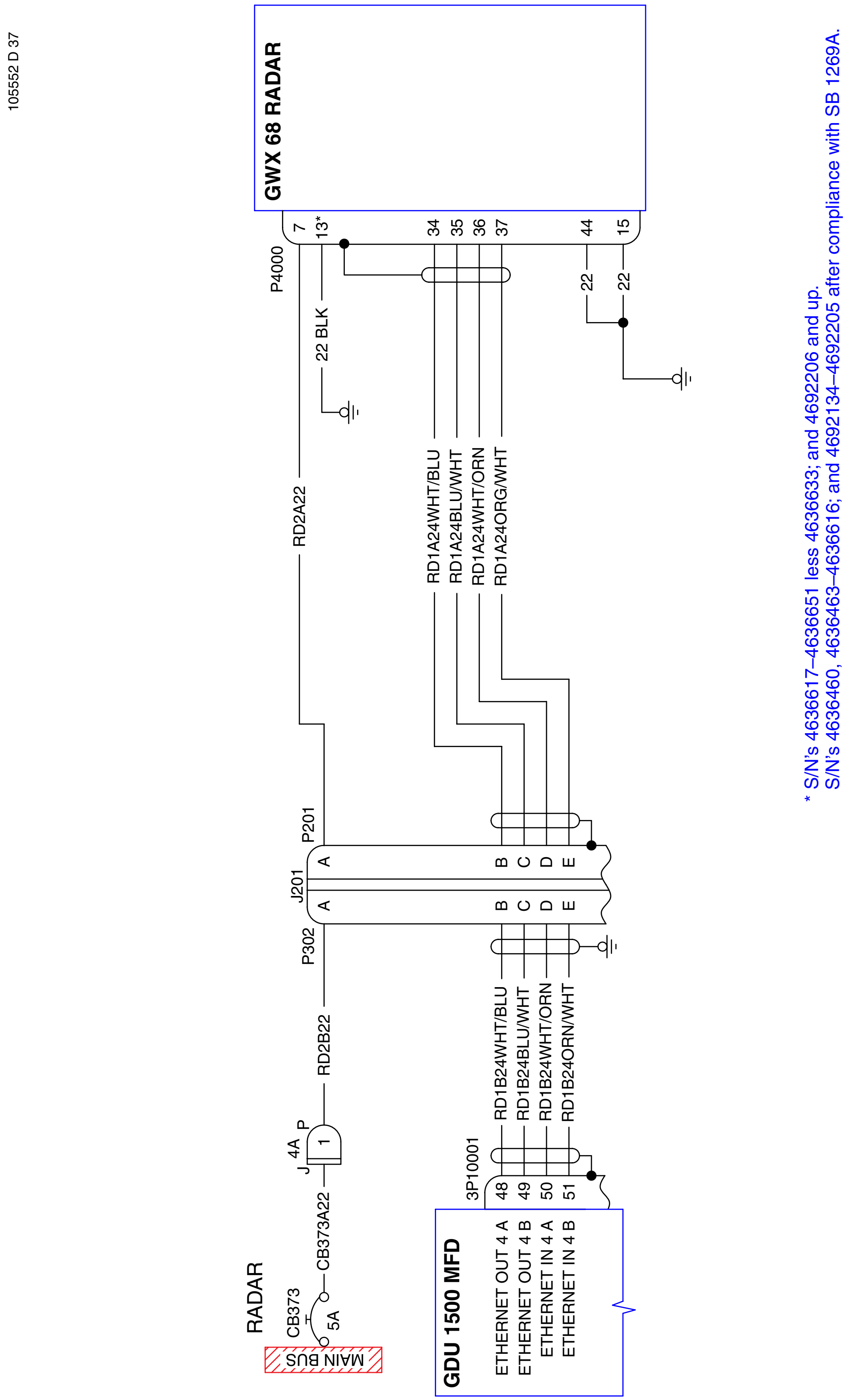
* not with G1000



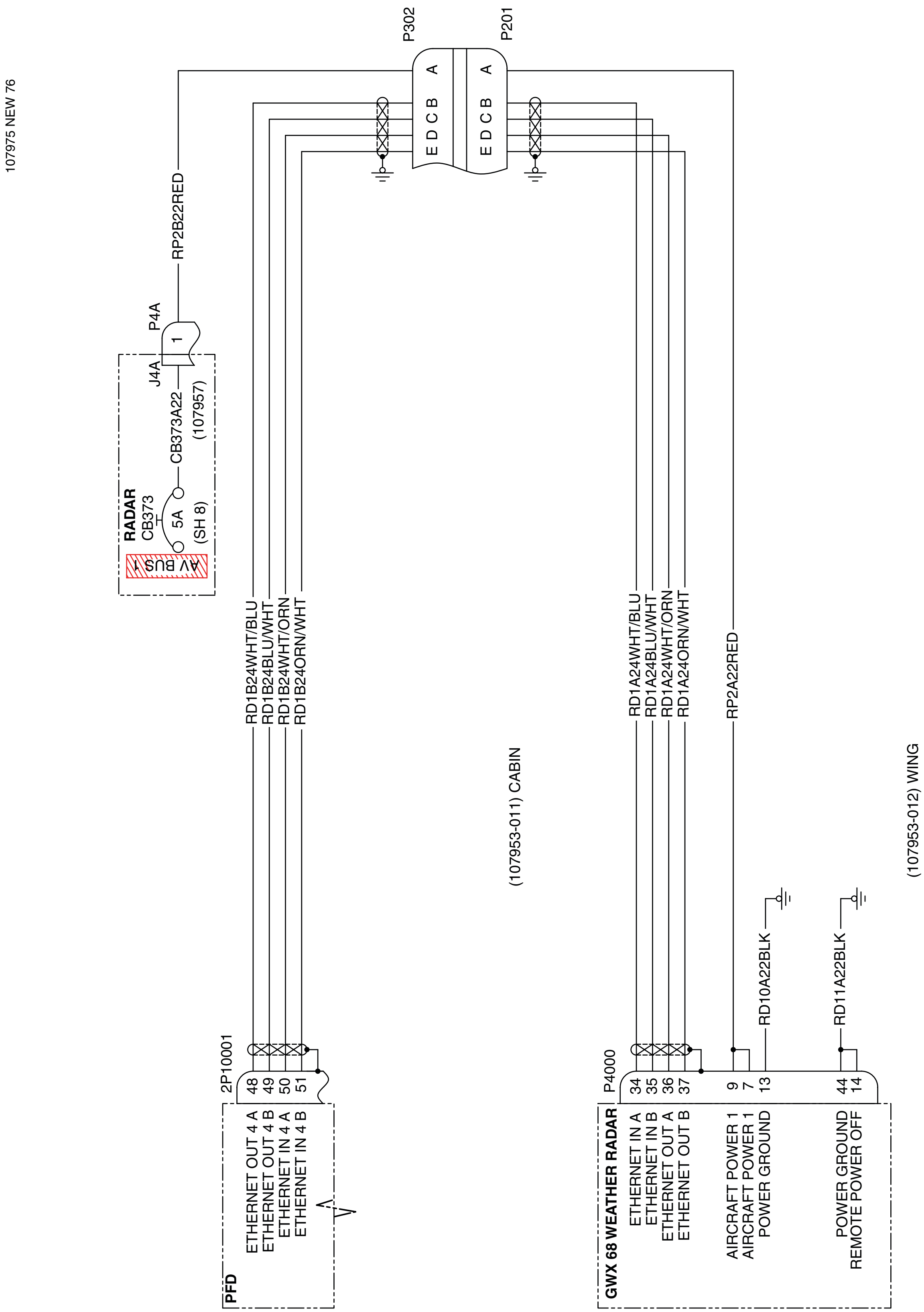
Weather Radar
Figure 1 (Sheet 1 of 4)



Weather Radar
Figure 1 (Sheet 2 of 4)



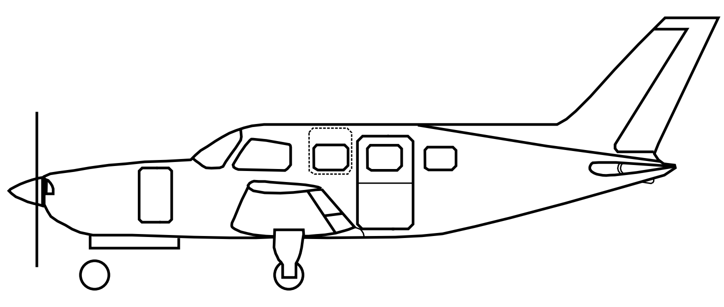
Weather Radar
Figure 1 (Sheet 3 of 4)



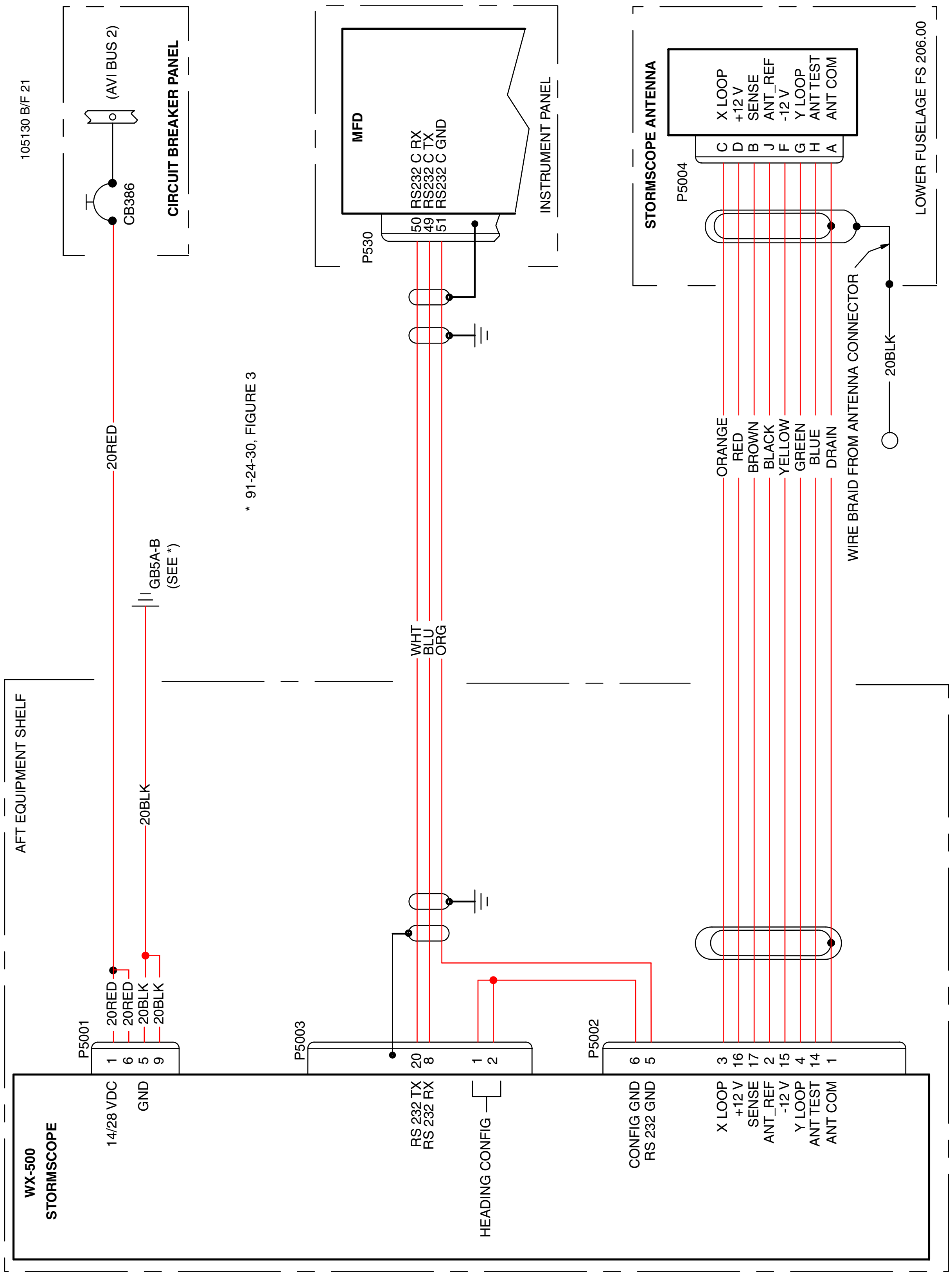
Weather Radar
Figure 1 (Sheet 4 of 4)

Item #	Designation	Description

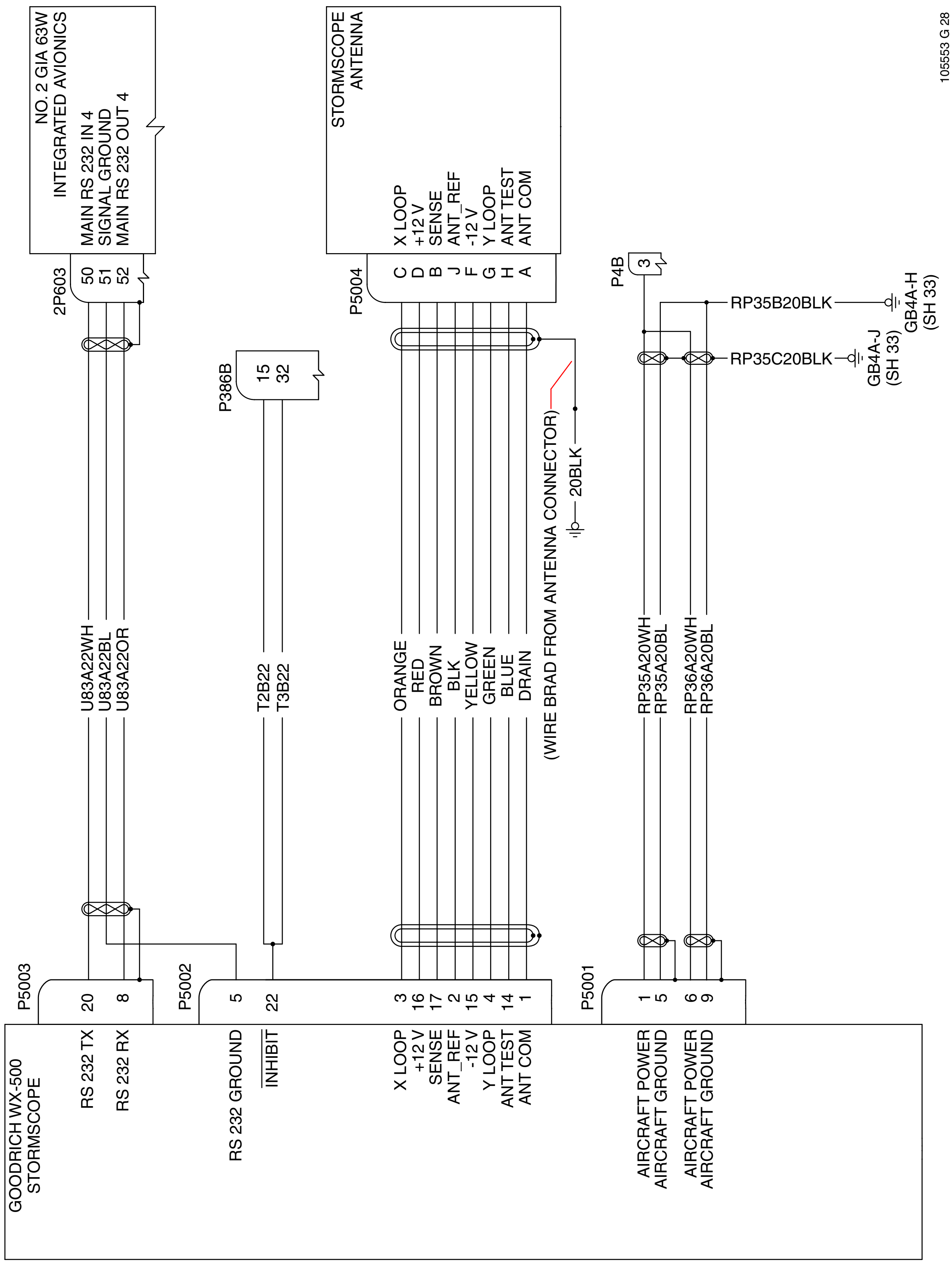
Information Pending - See Parts Catalog



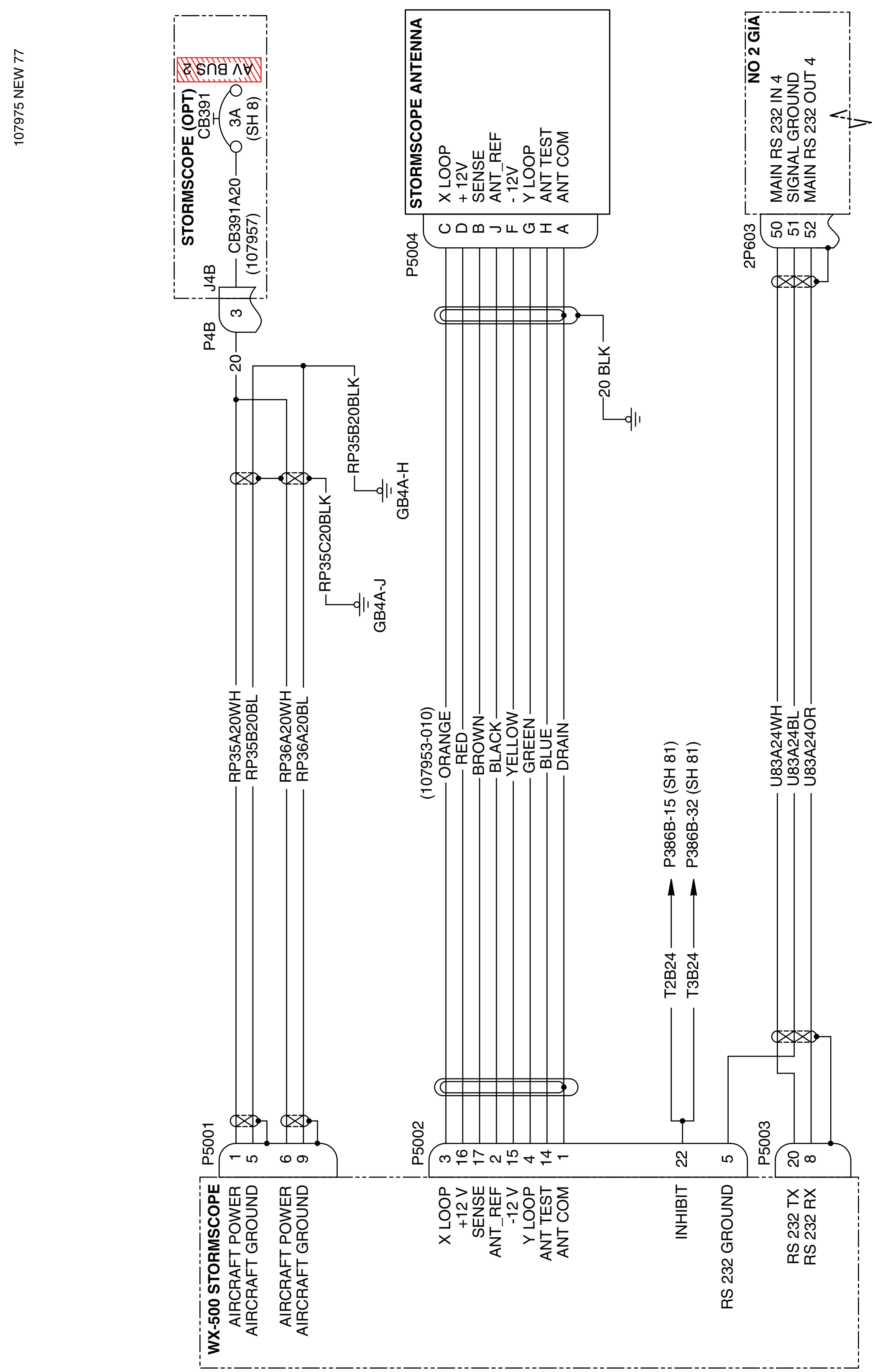
WX-500 Stormscope
Figure 2 (Sheet 1 of 5)



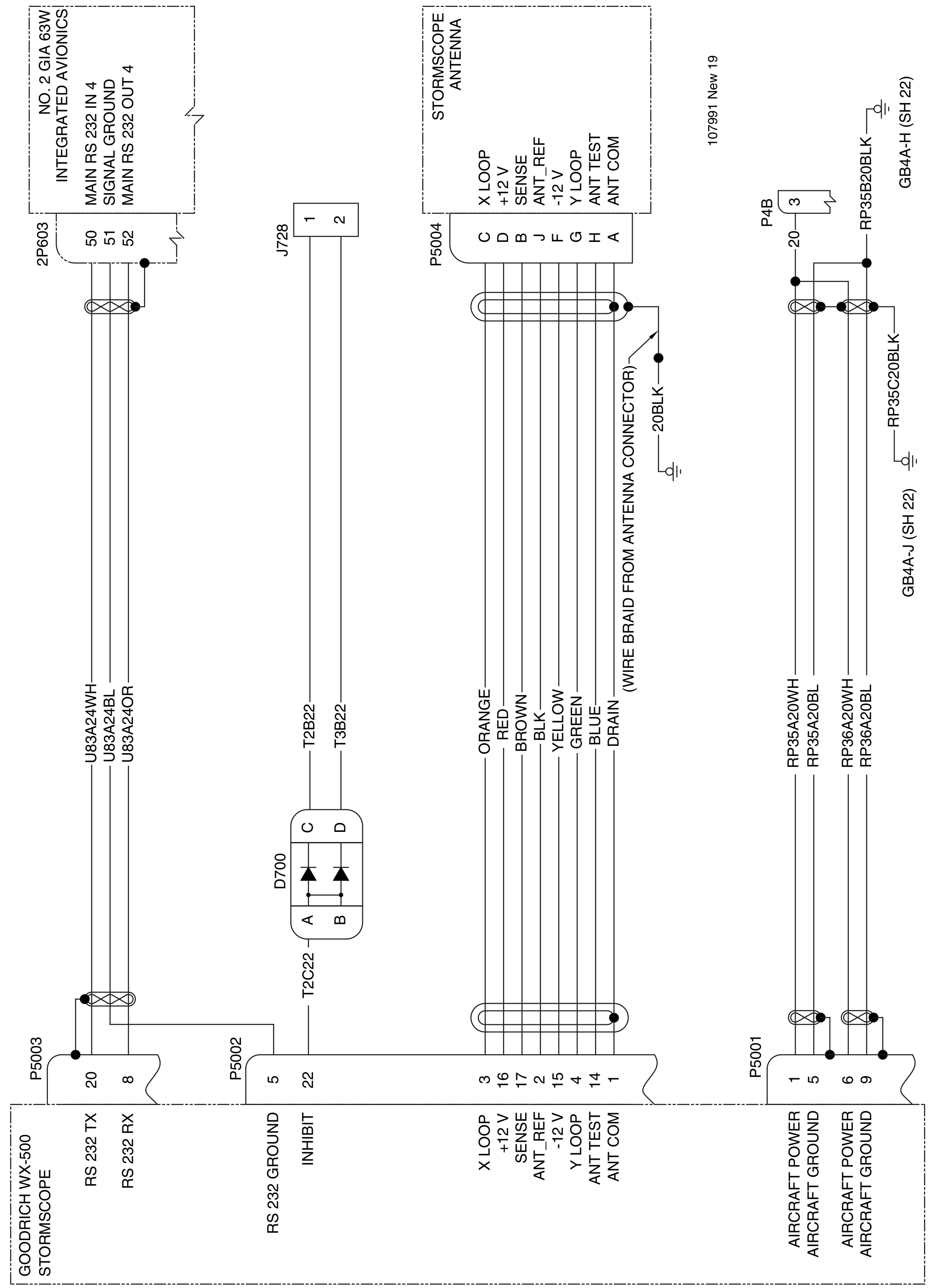
WX-500 Stormscope
Figure 2 (Sheet 2 of 5)



WX-500 Stormscope
Figure 2 (Sheet 3 of 5)



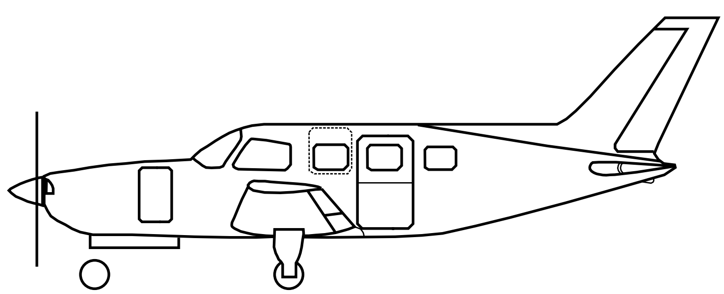
WX-500 Stormscope
Figure 2 (Sheet 4 of 5)



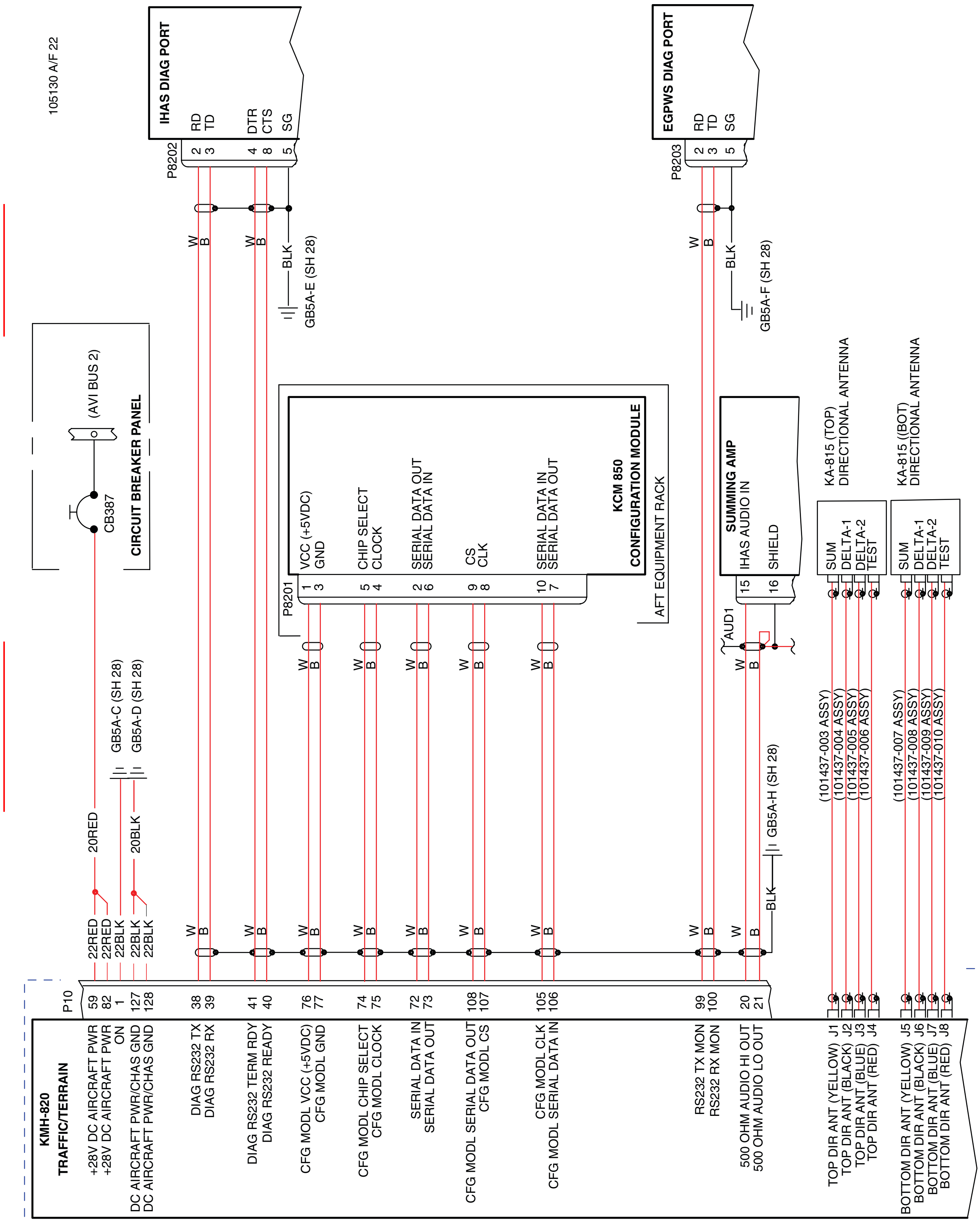
WX-500 Stormscope
Figure 2 (Sheet 5 of 5)

Item #	Designation	Description

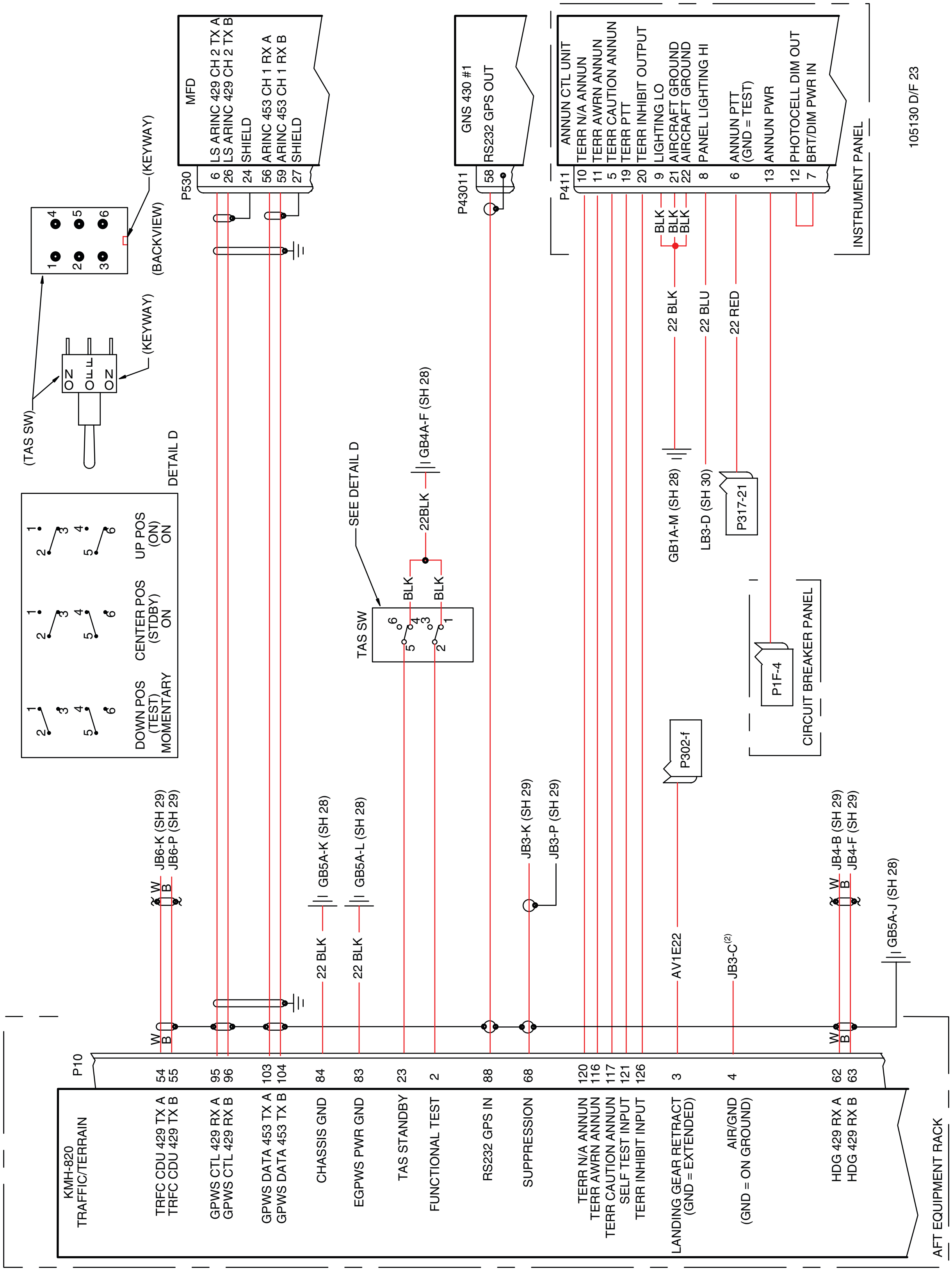
Information Pending - See Parts Catalog



Integrated Hazard Alert System (IHAS)
Figure 3 (Sheet 1 of 3)



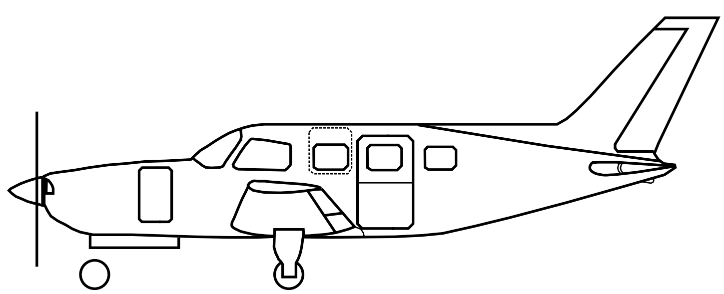
Integrated Hazard Alert System (IHAS)
Figure 3 (Sheet 2 of 3)



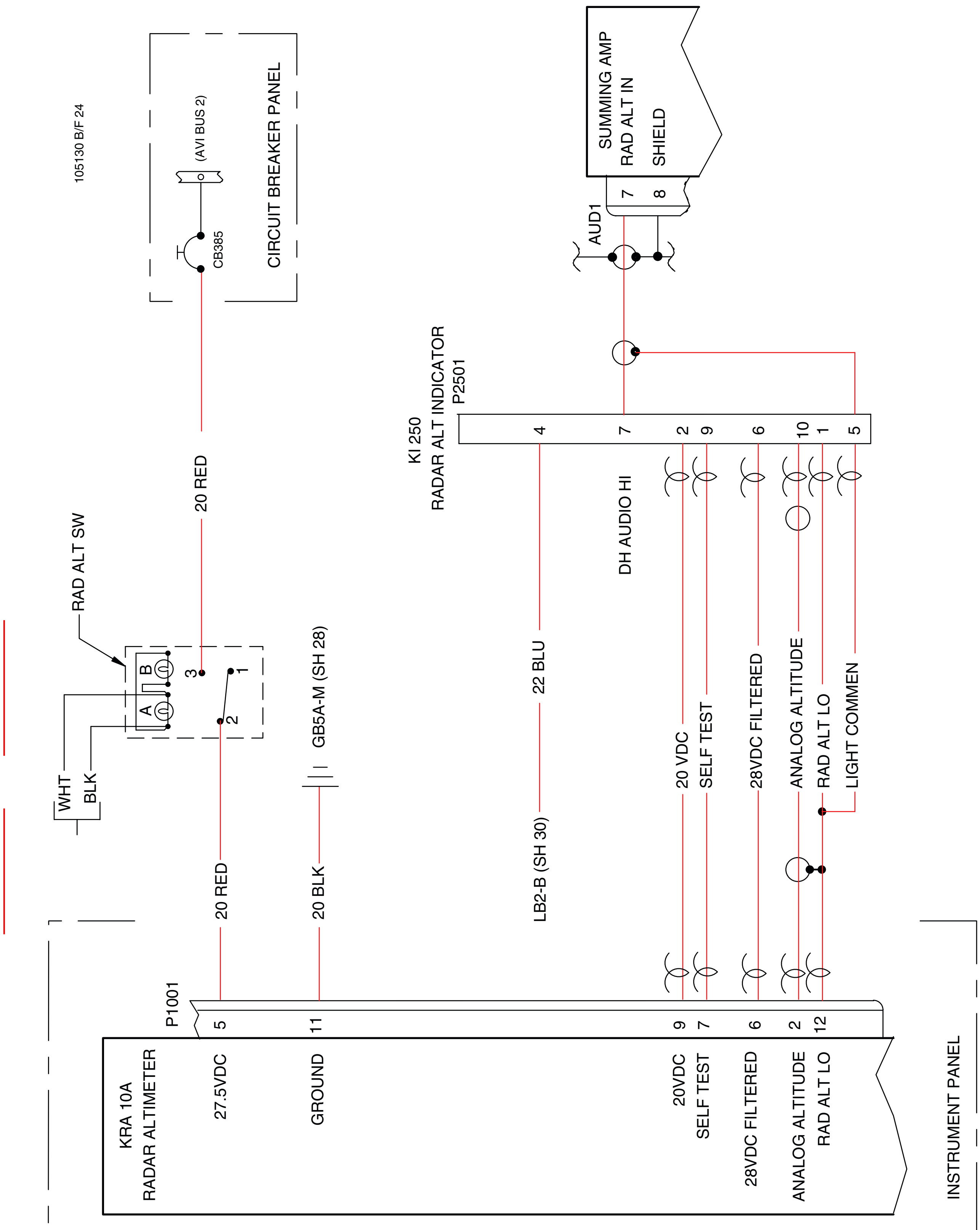
Integrated Hazard Alert System (IHAS)
Figure 3 (Sheet 3 of 3)

Item #	Designation	Description

Information Pending - See Parts Catalog



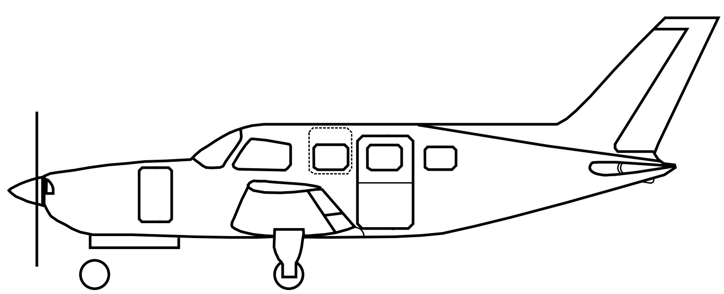
KRA 10A Radar Altimeter
Figure 4 (Sheet 1 of 2)



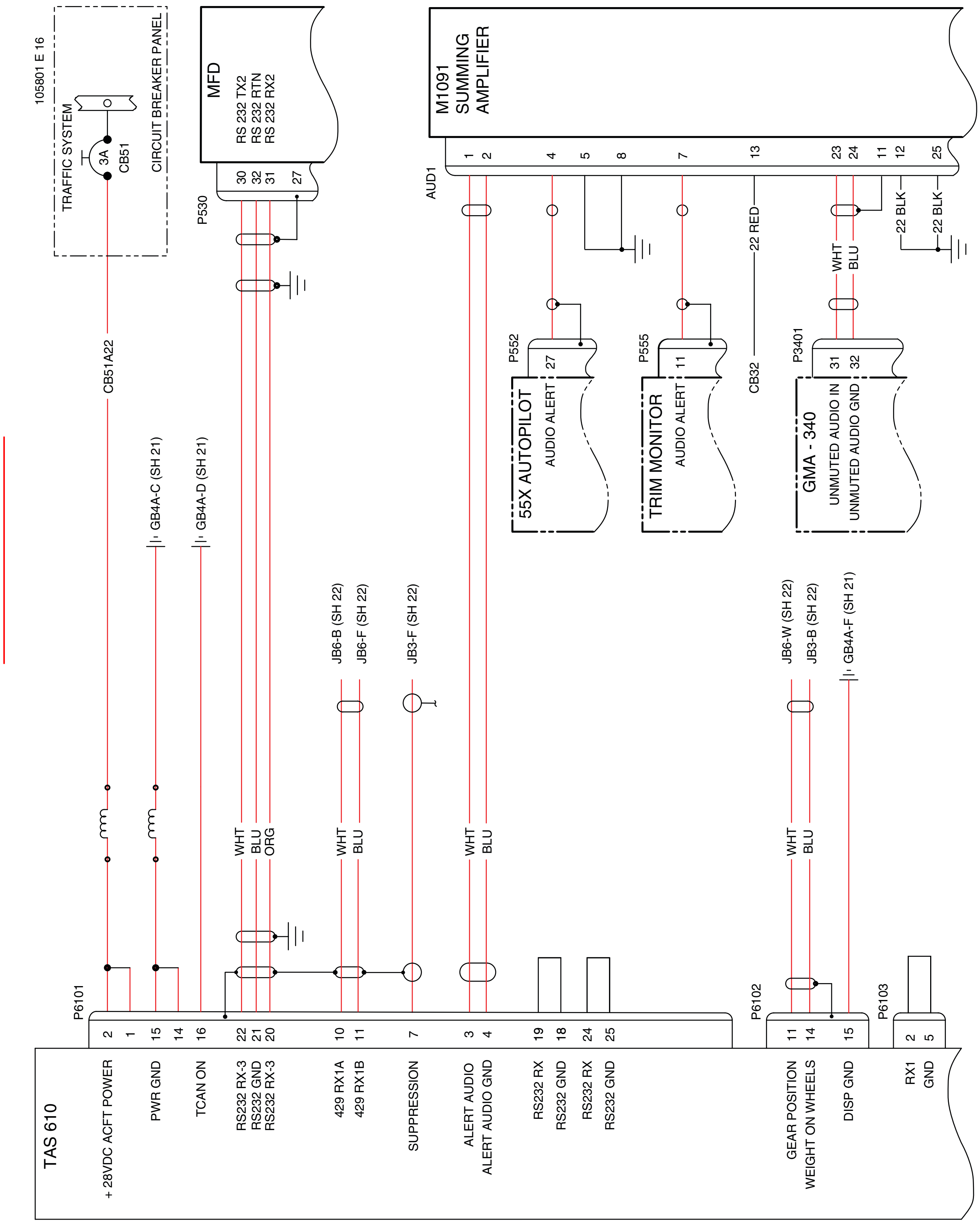
KRA 10A Radar Altimeter
Figure 4 (Sheet 2 of 2)

Item #	Designation	Description

Information Pending - See Parts Catalog



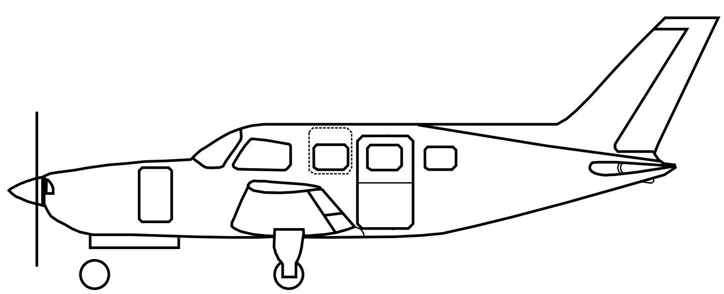
TAS-610 Traffic Advisory System
Figure 5 (Sheet 1 of 2)



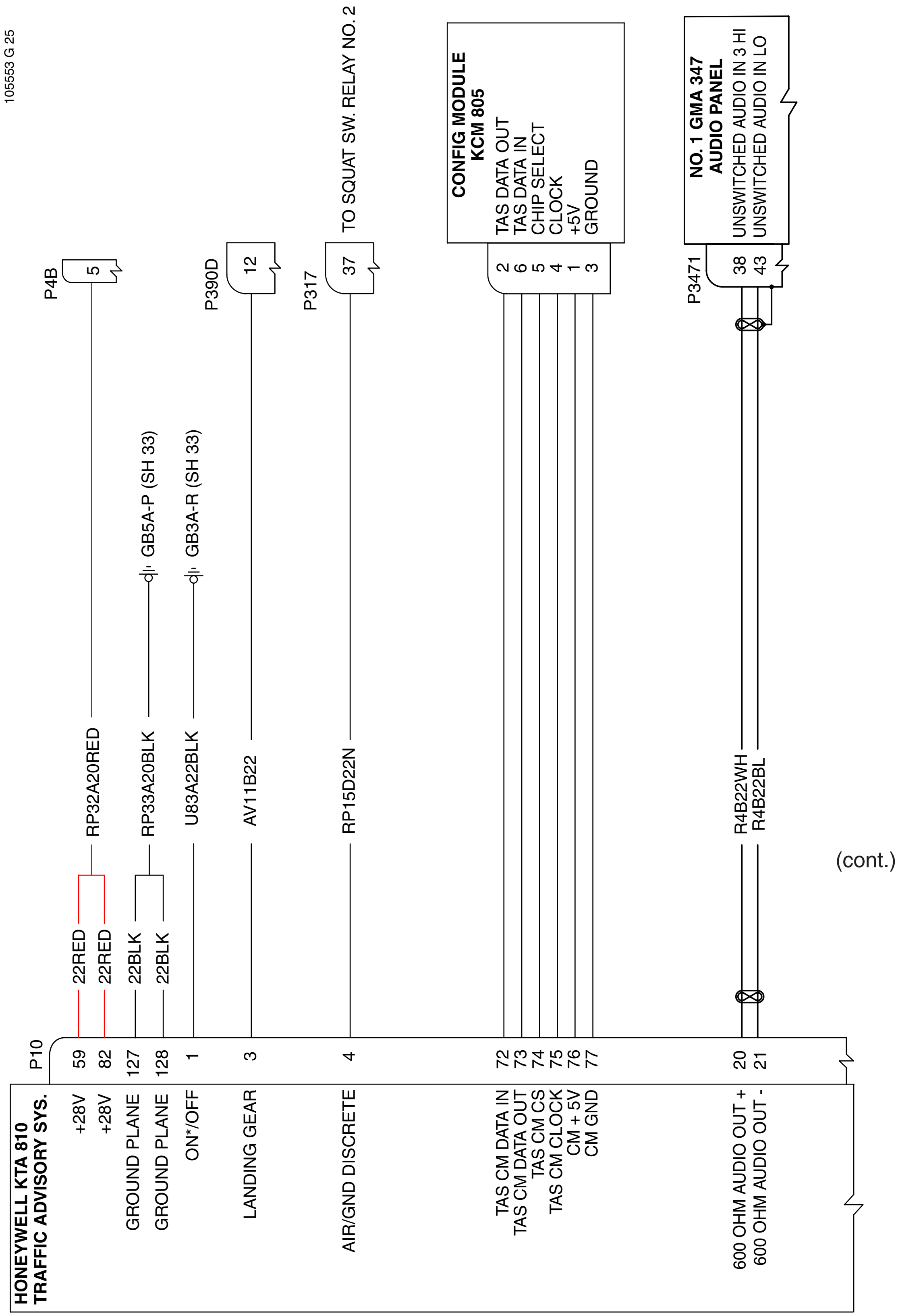
TAS-610 Traffic Advisory System
Figure 5 (Sheet 2 of 2)

Item #	Designation	Description

Information Pending - See Parts Catalog



KTA-810 Traffic Advisory System
Figure 6 (Sheet 1 of 3)



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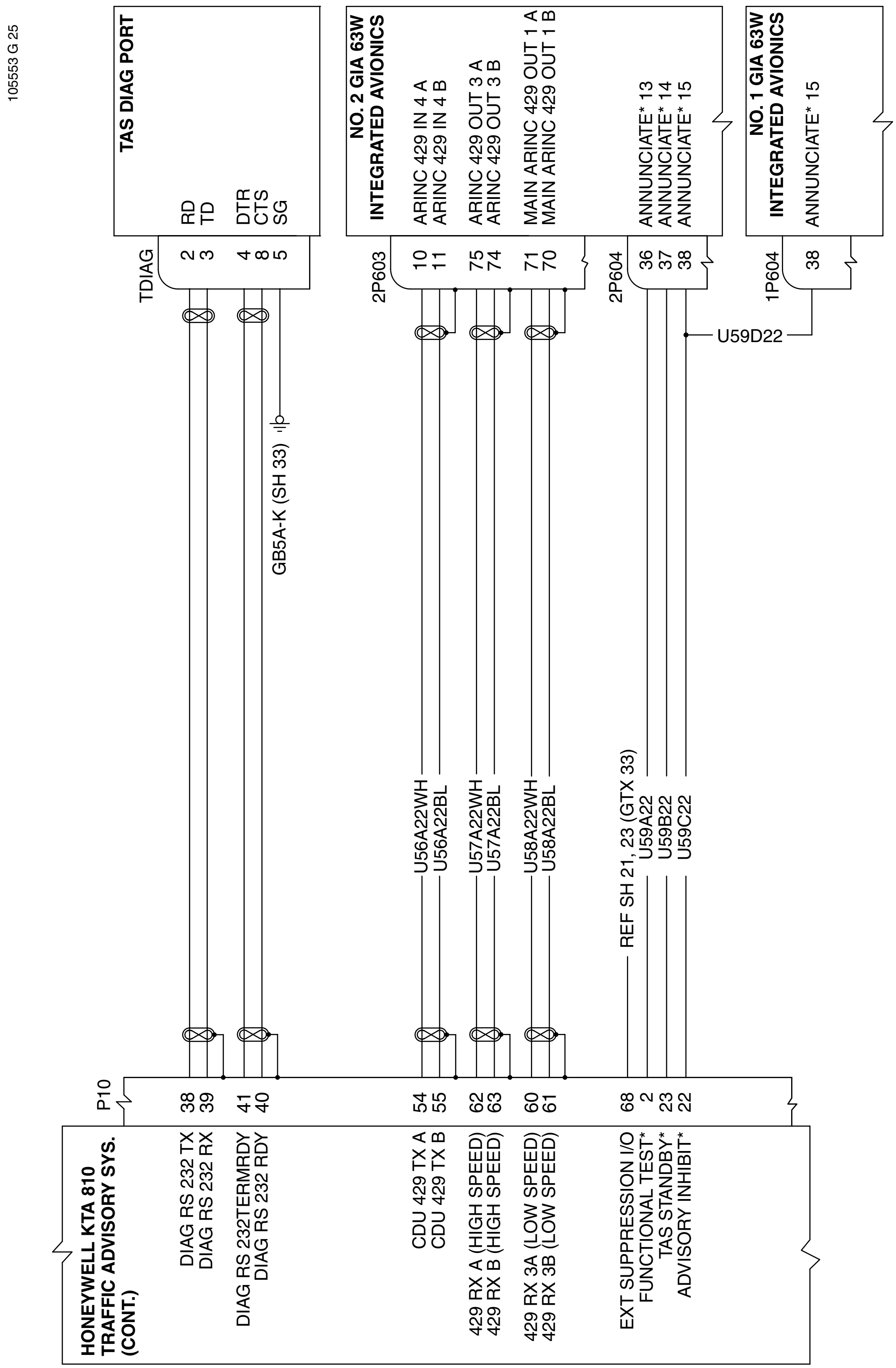
KTA-810 Traffic Advisory System
Figure 6 (Sheet 2 of 3)



KTA-810 Traffic Advisory System
Figure 6 (Sheet 3 of 3)

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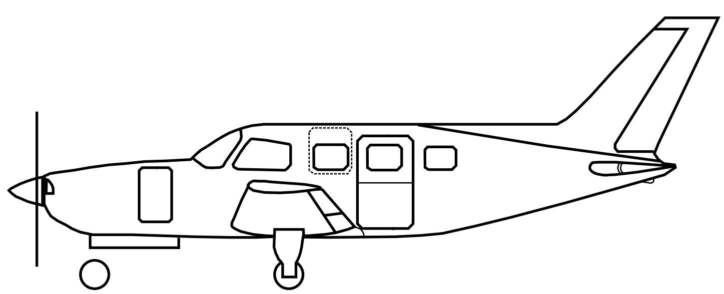
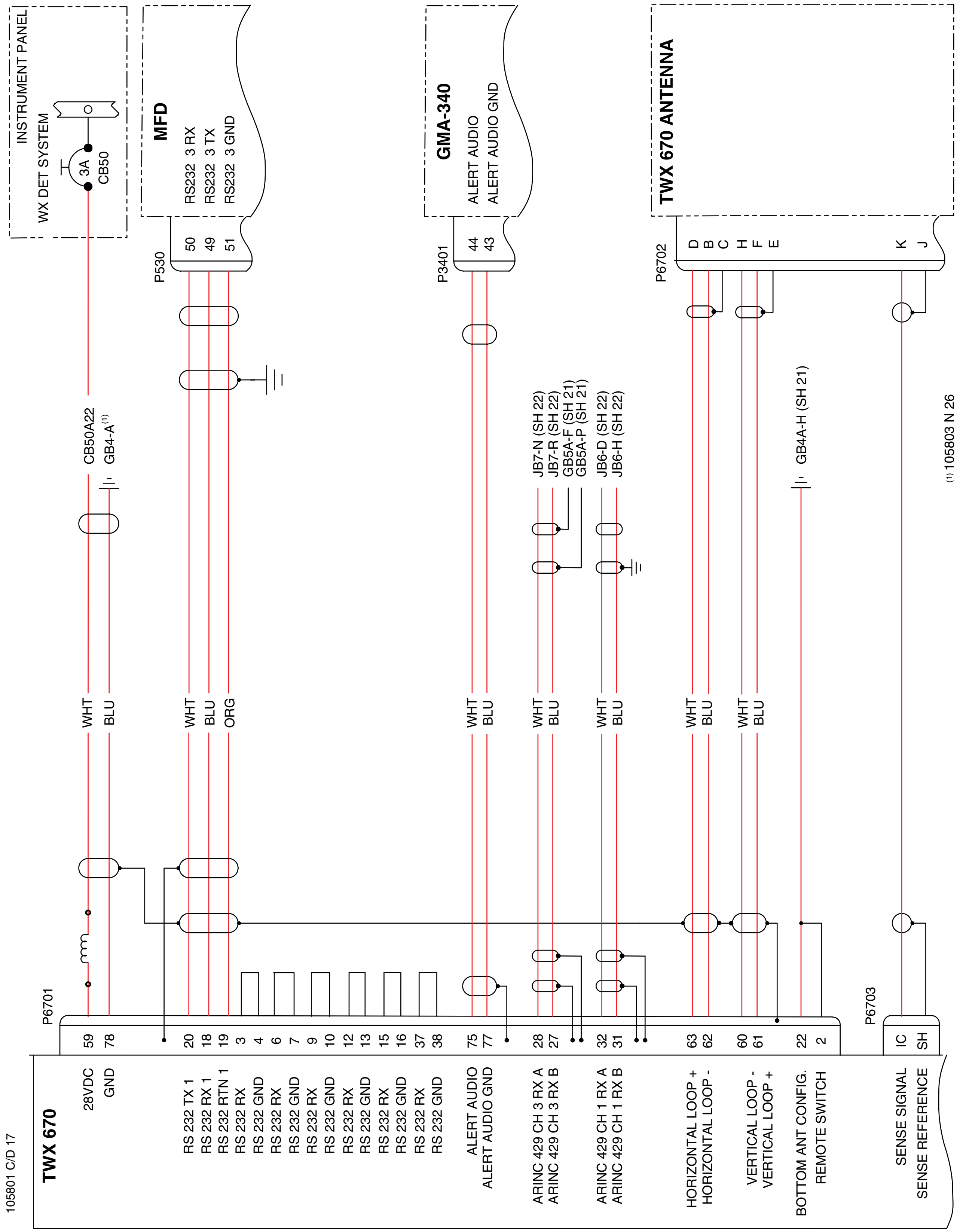
Item #	Designation	Description

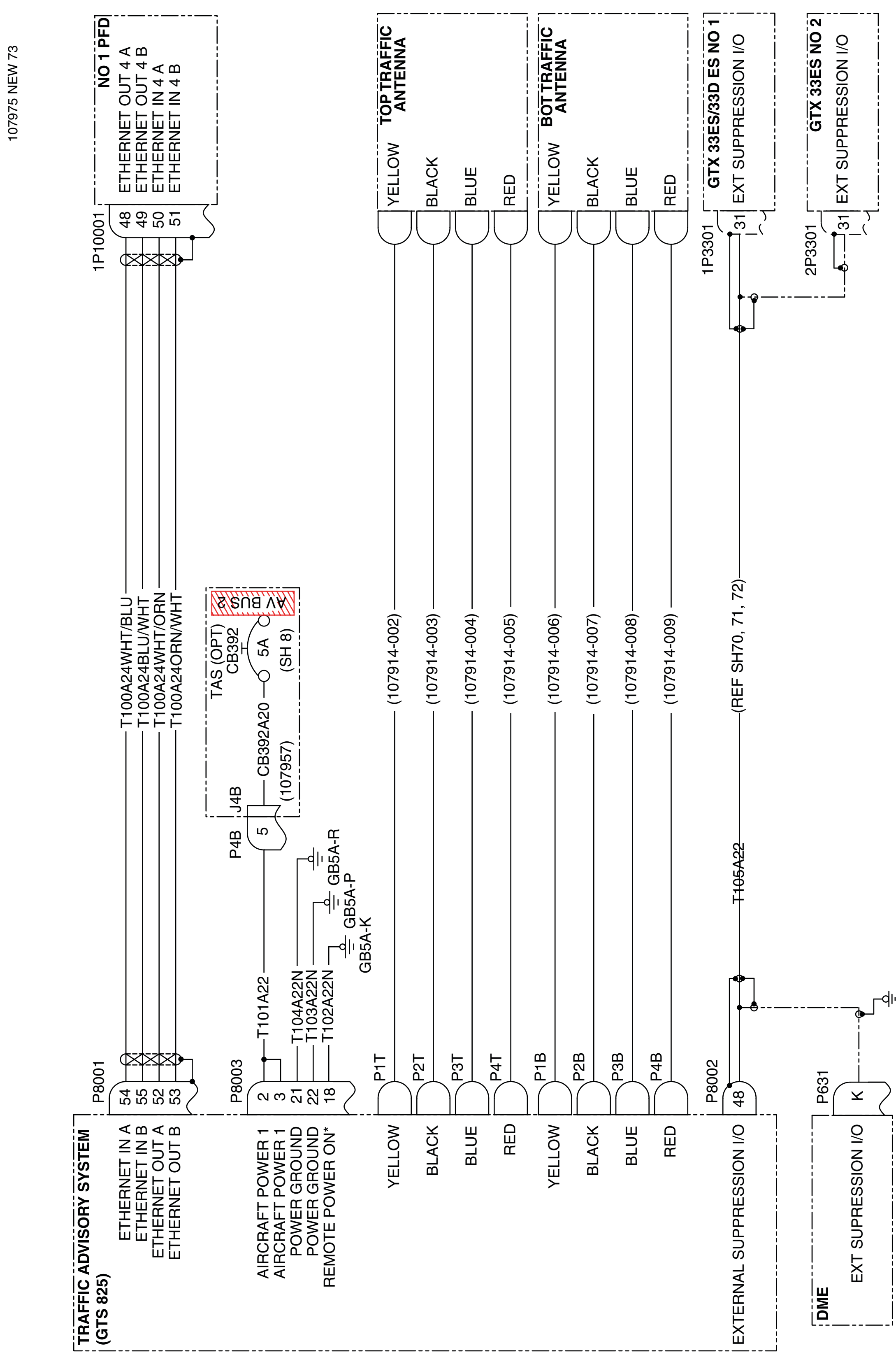
Information Pending - See Parts Catalog



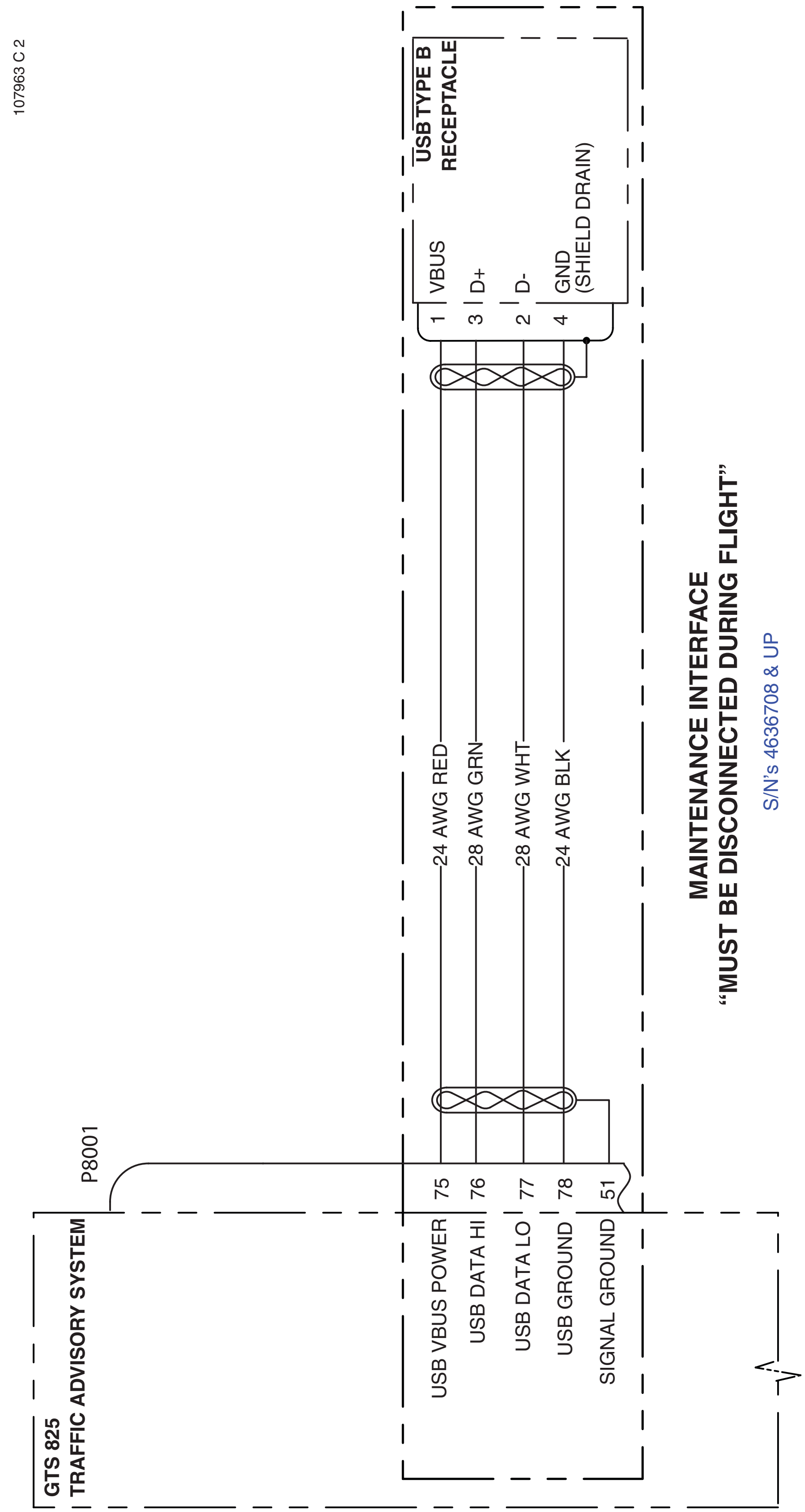
TWX-670 Weather Detection System
Figure 7 (Sheet 1 of 2)



TWX-670 Weather Detection System
Figure 7 (Sheet 2 of 2)

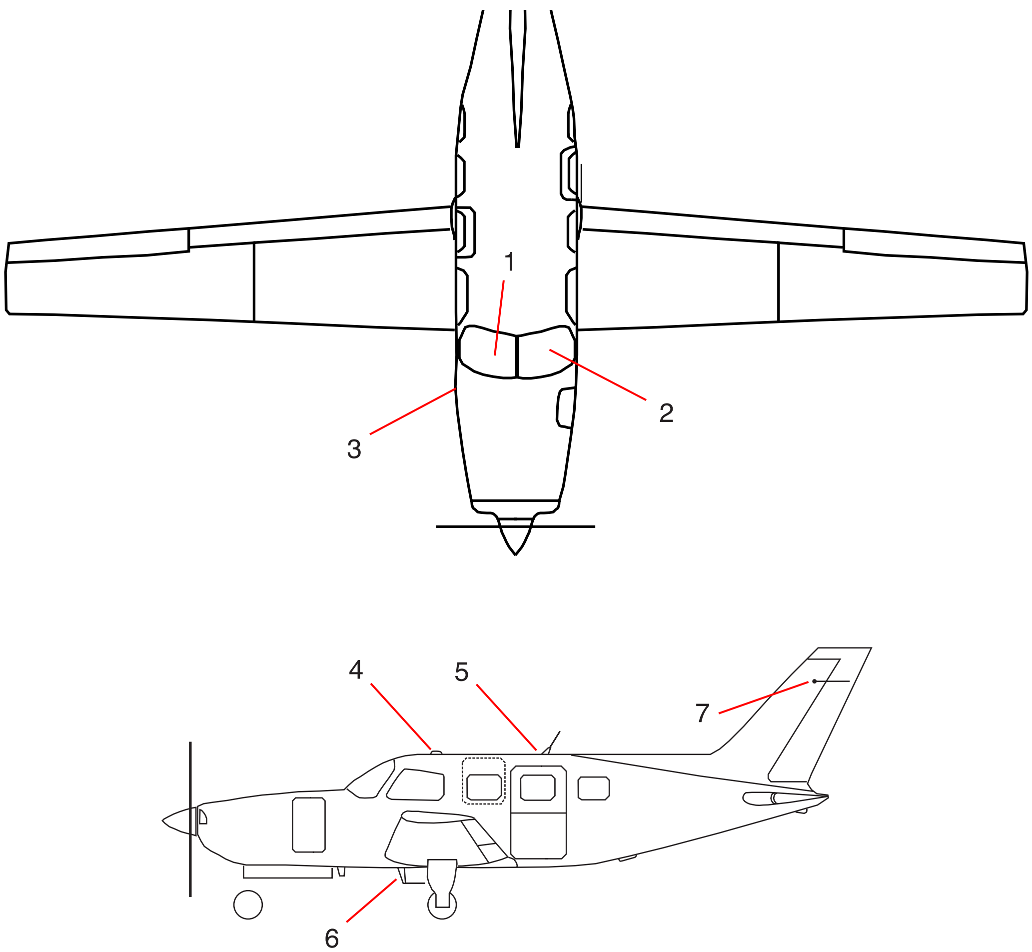


GTS 825 Traffic Advisory System (Optional)
Figure 8 (Sheet 1 of 2)

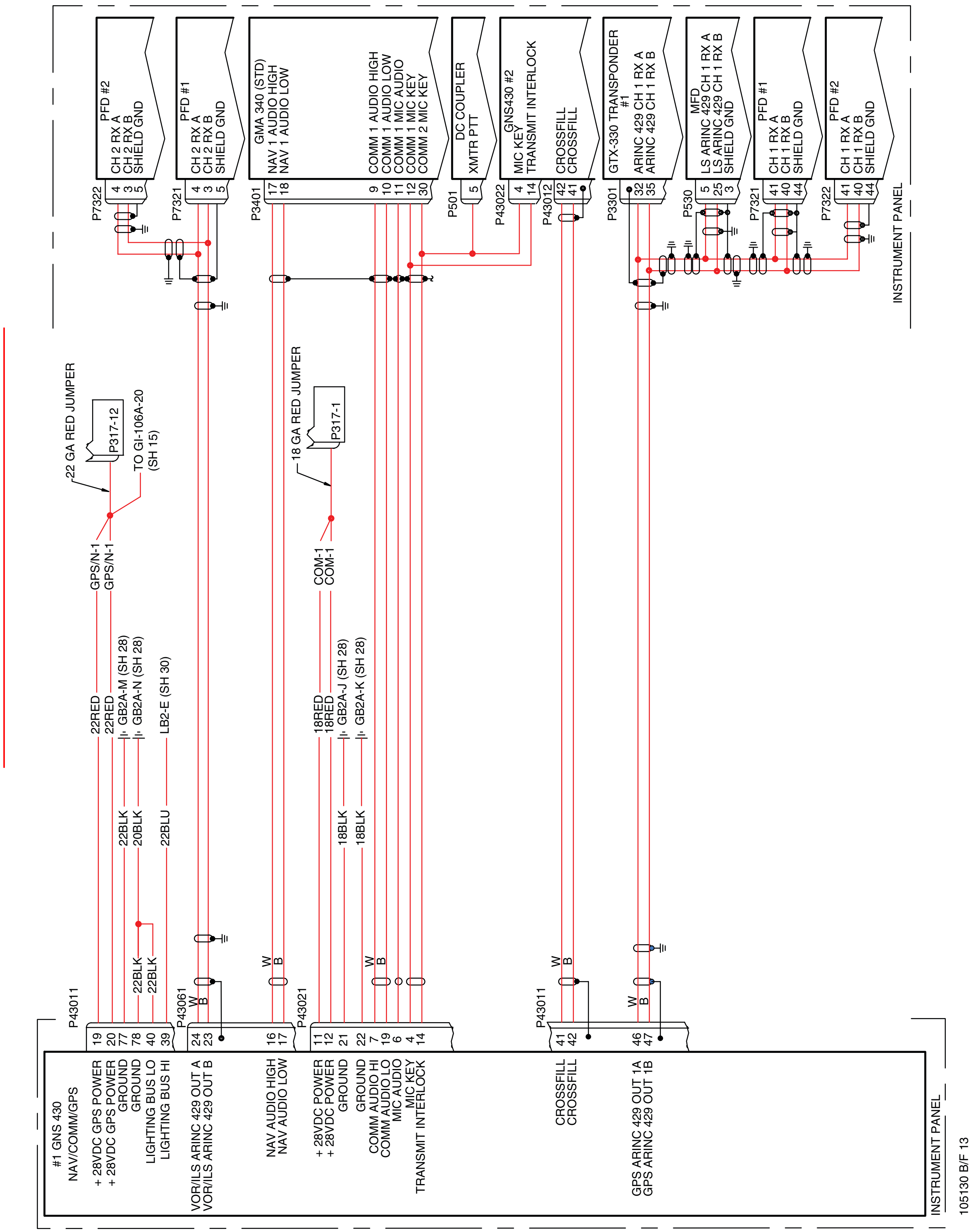


GTS 825 Traffic Advisory System (Optional)
Figure 8 (Sheet 2 of 2)

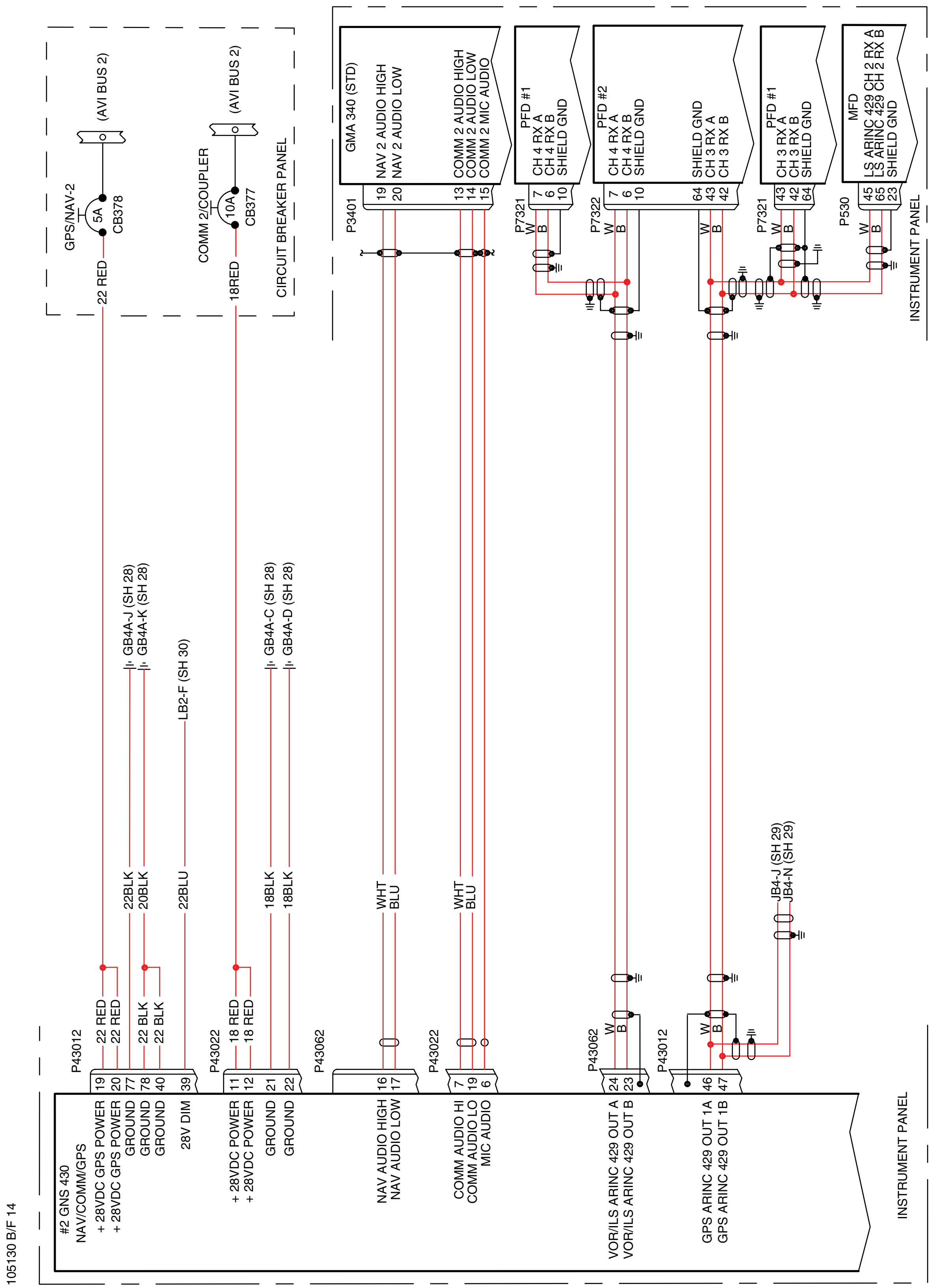
Item #	Designation	Description
1	Garmin GNS-430	Global Navigation System
2	Garmin GI-106	Course Deviation (NAV) Indicator
3		NAV Antenna Coupler
4		GPS#1 Antenna (i.e. -left side)
		GPS#2 Antenna (i.e. -right side)
5		COM#1 Antenna
6		COM#2 Antenna
7		NAV Antenna



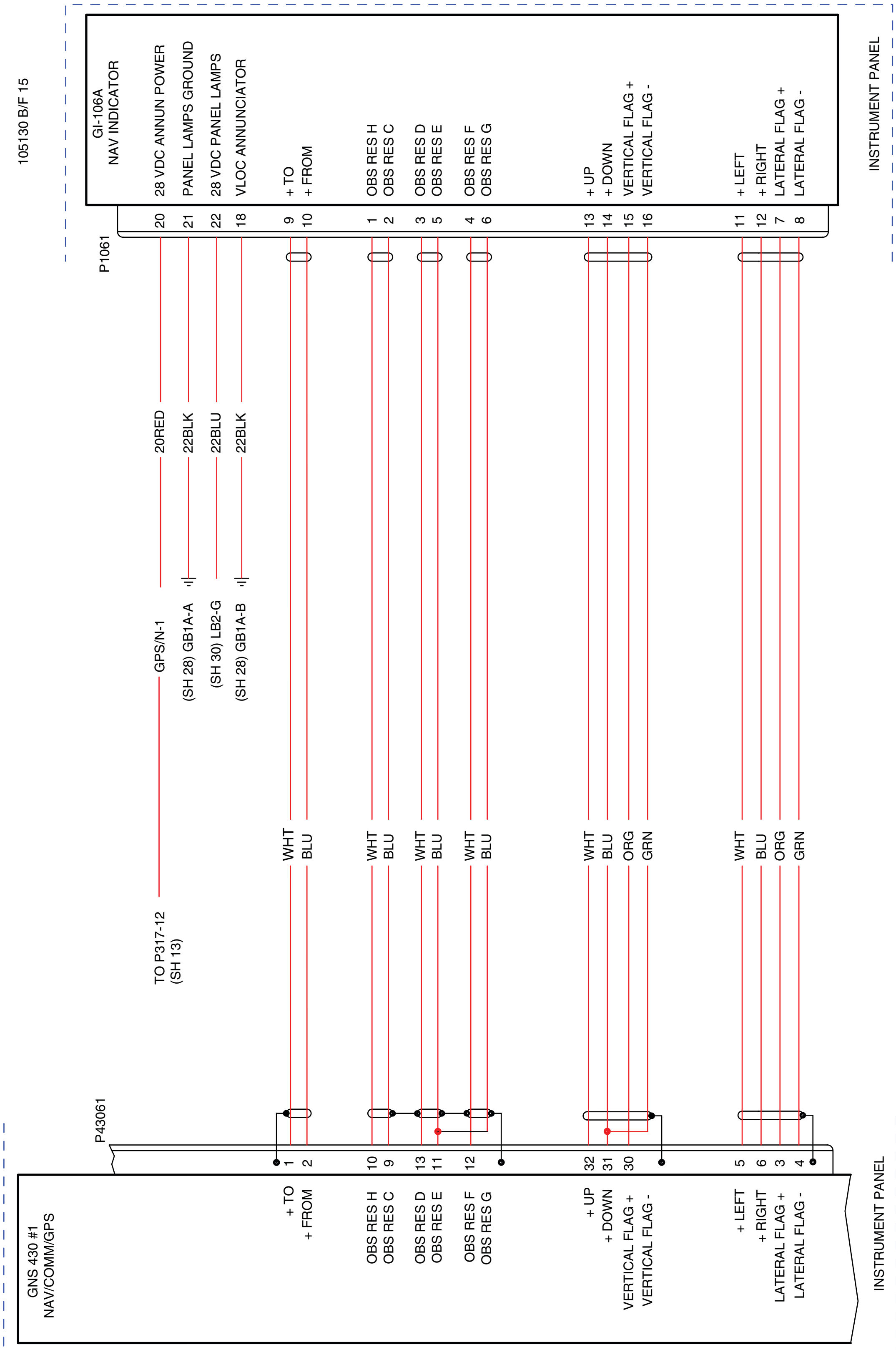
GNS-430 - COM/NAV/GPS
Figure 1 (Sheet 1 of 7)



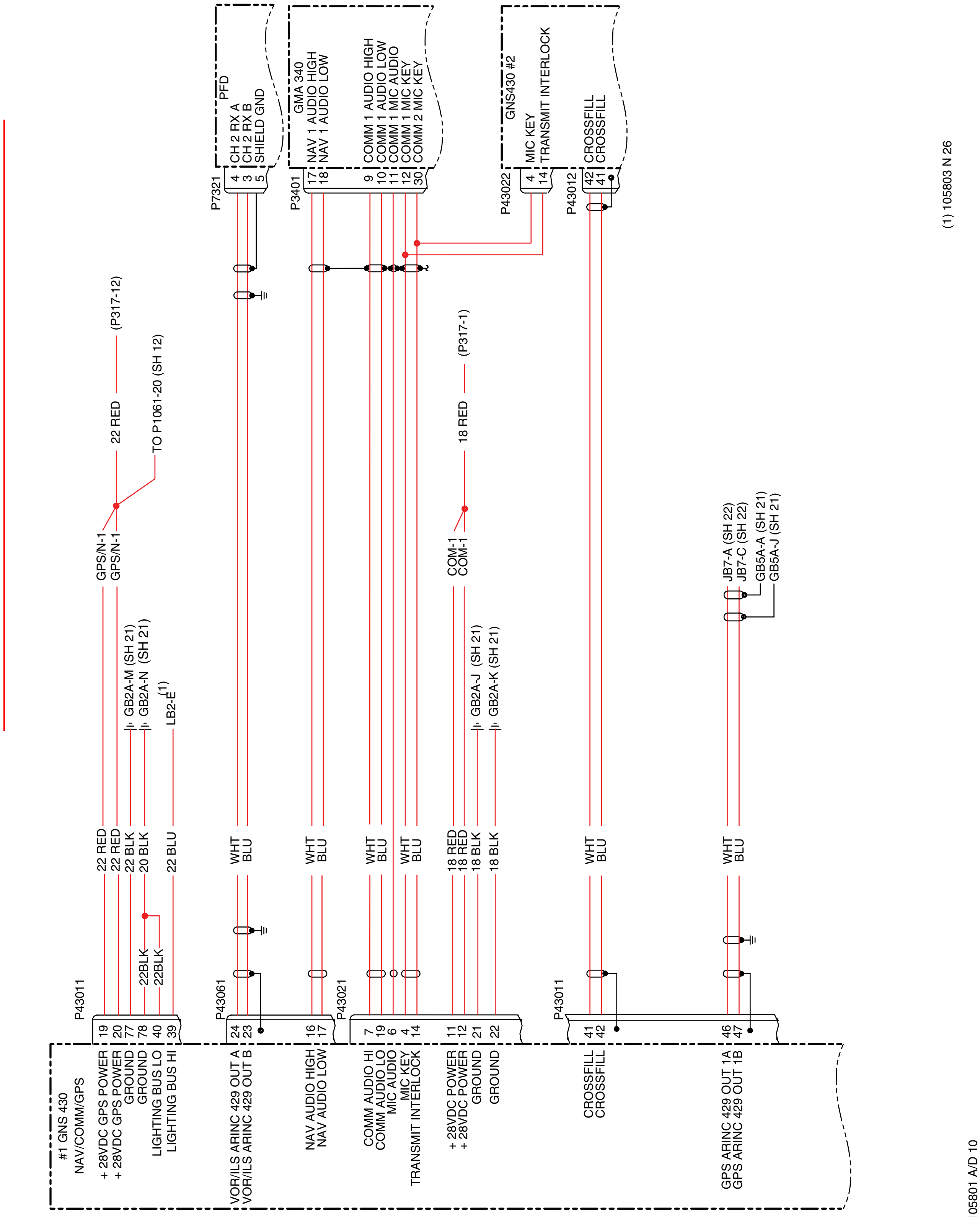
GNS 430 - COM/NAV/GPS
Figure 1 (Sheet 2 of 7)



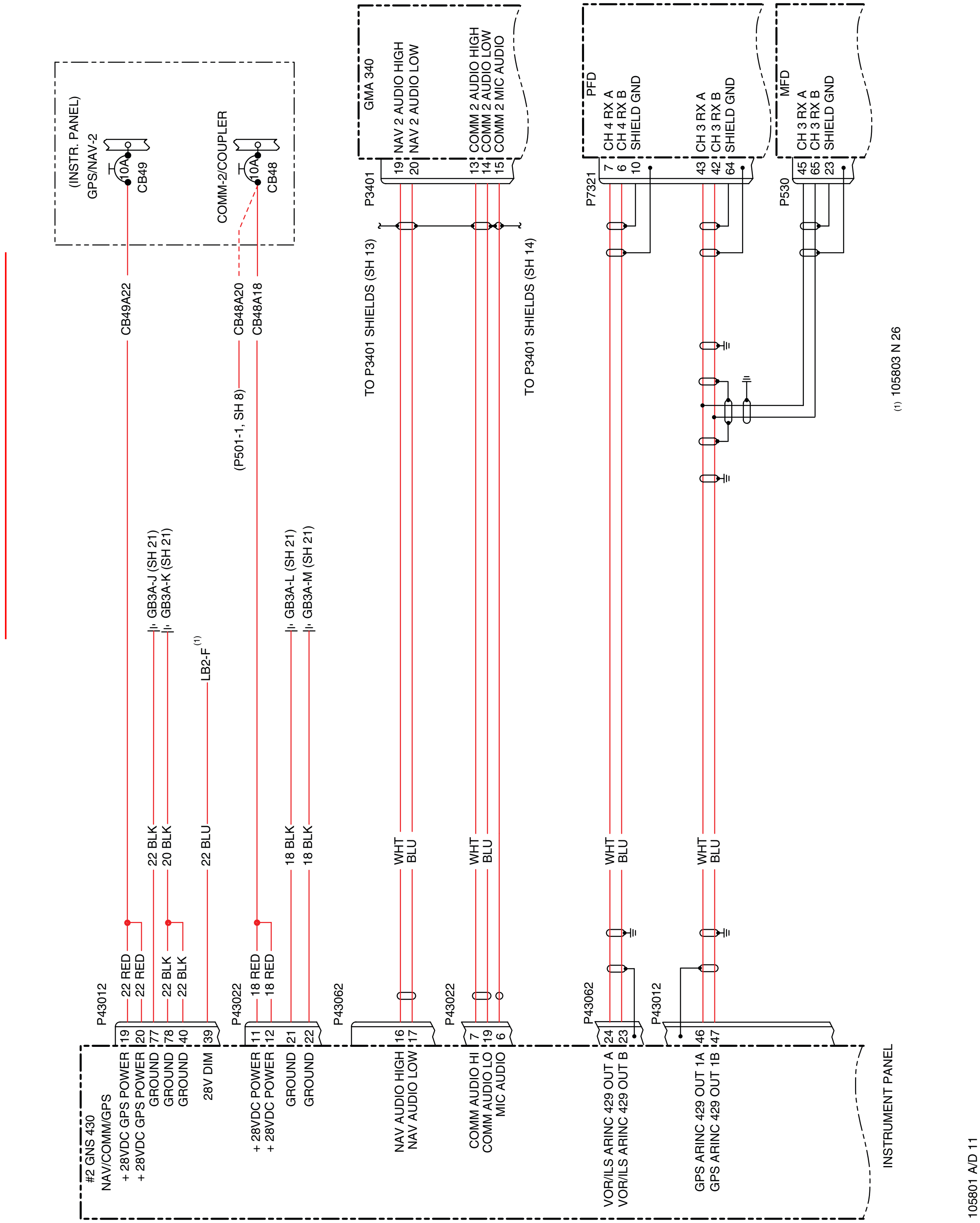
GNS 430 - COM/NAV/GPS
Figure 1 (Sheet 3 of 7)



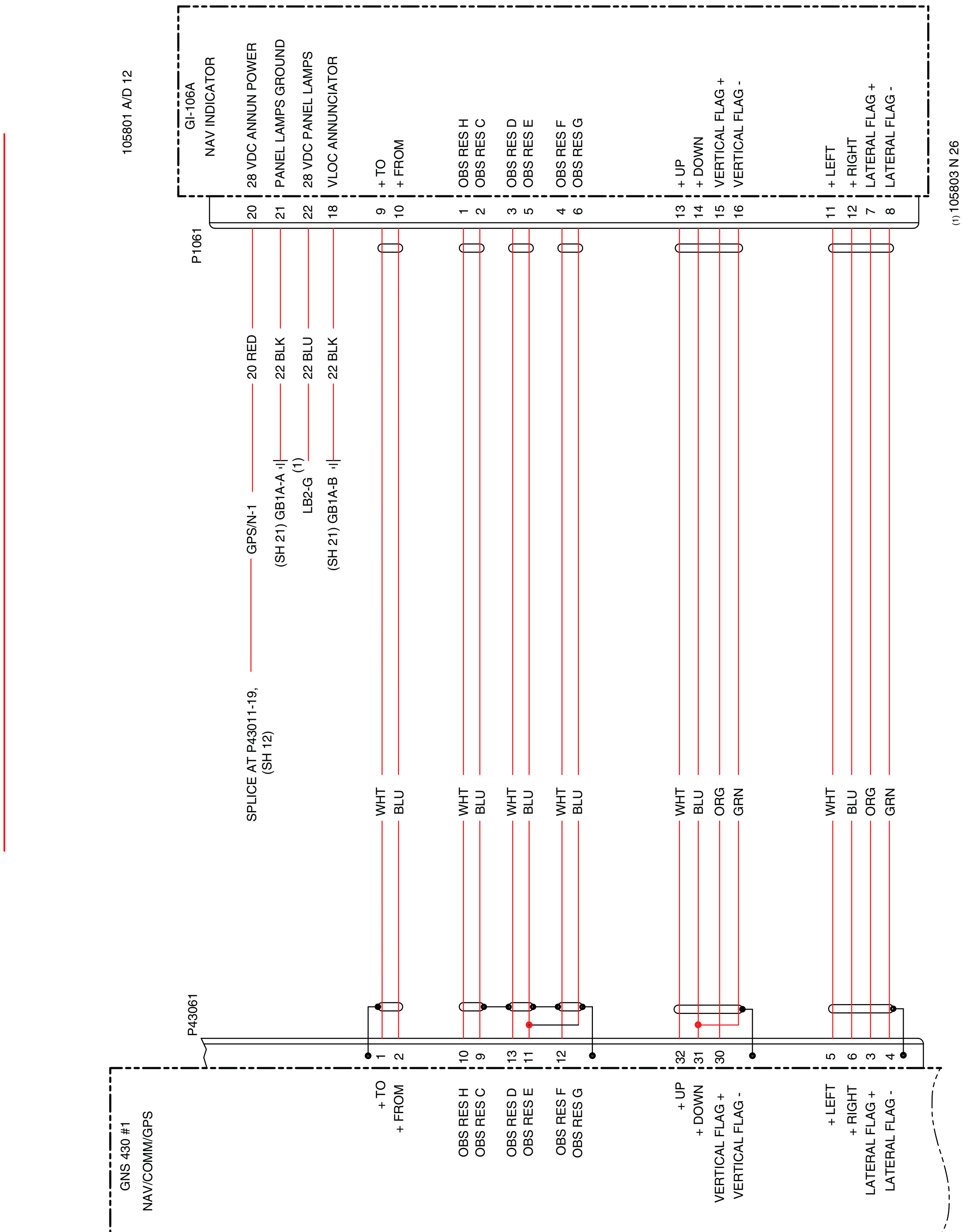
GNS 430 - COM/NAV/GPS
Figure 1 (Sheet 4 of 7)



GNS 430 - COM/NAV/GPS
Figure 1 (Sheet 5 of 7)



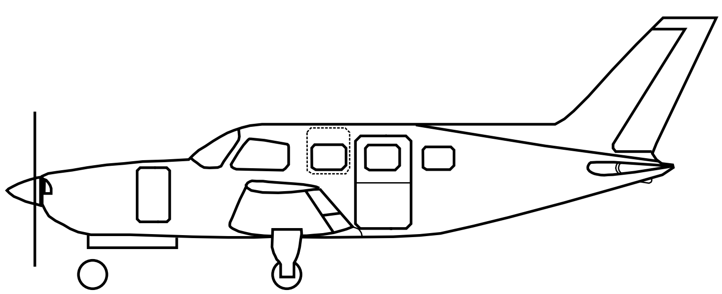
GNS 430 - COM/NAV/GPS
Figure 1 (Sheet 6 of 7)



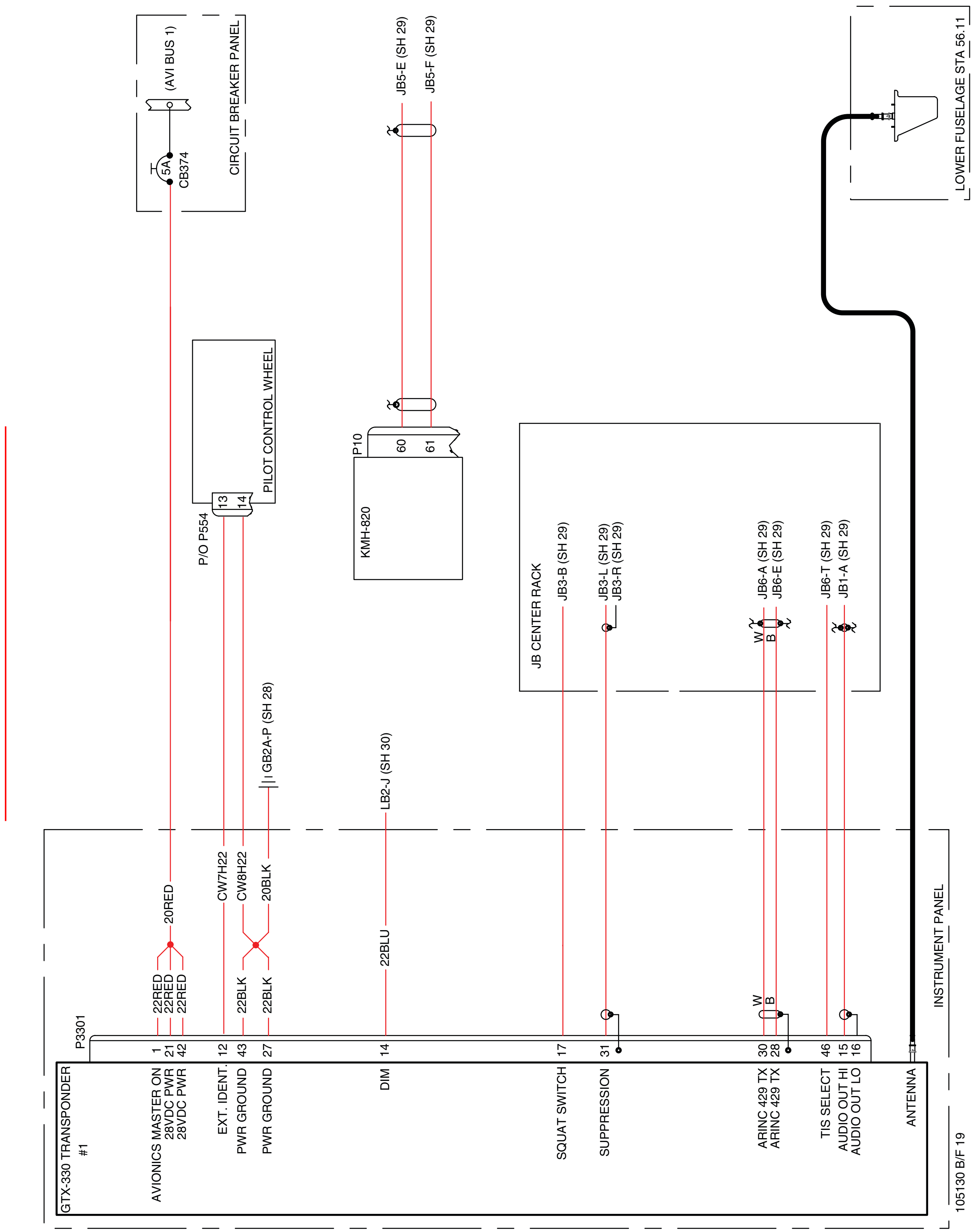
GNS 430 - COM/NAV/GPS
Figure 1 (Sheet 7 of 7)

Item #	Designation	Description

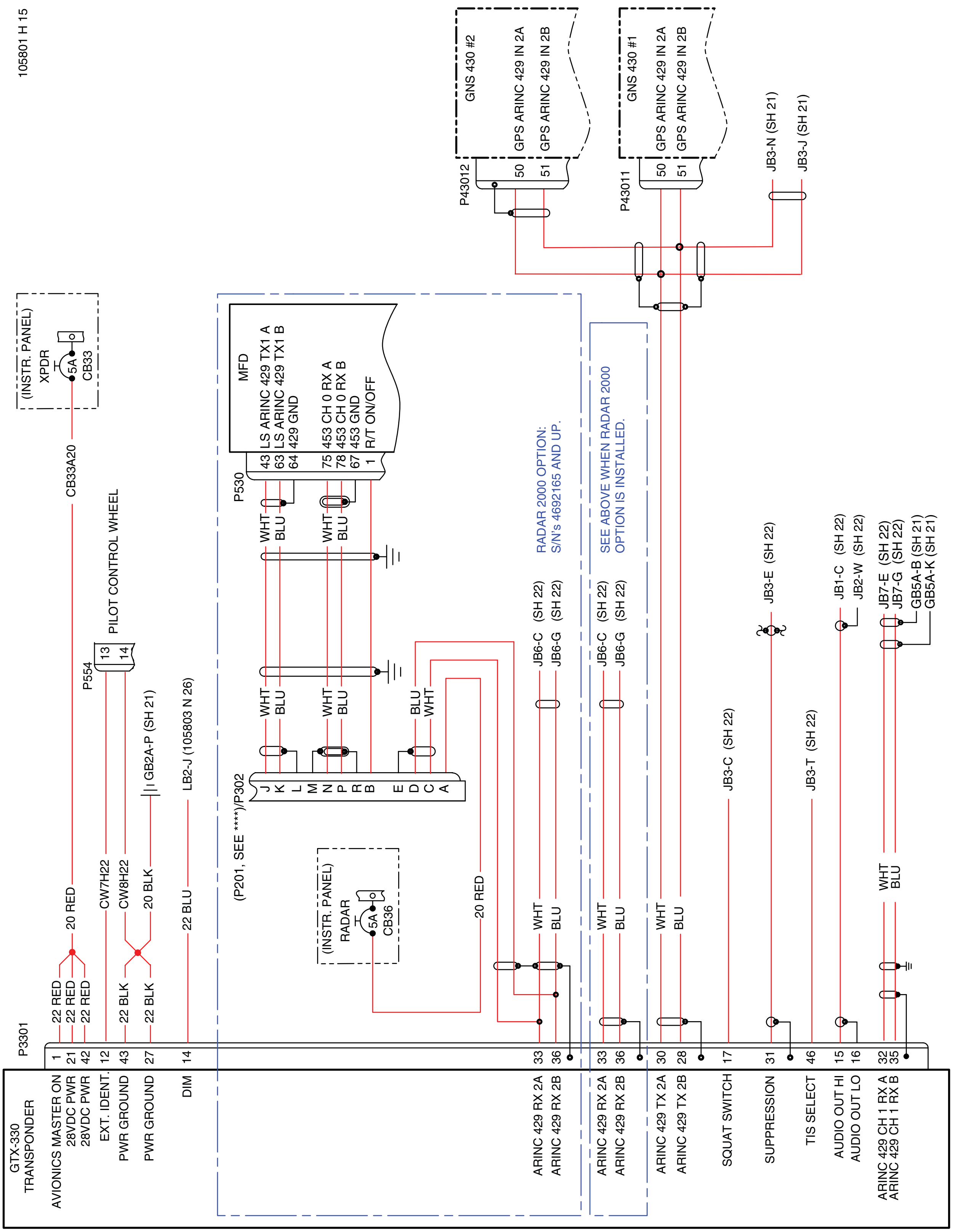
Information Pending - See Parts Catalog



GTX-330 - Single Transponder
Figure 2 (Sheet 1 of 3)



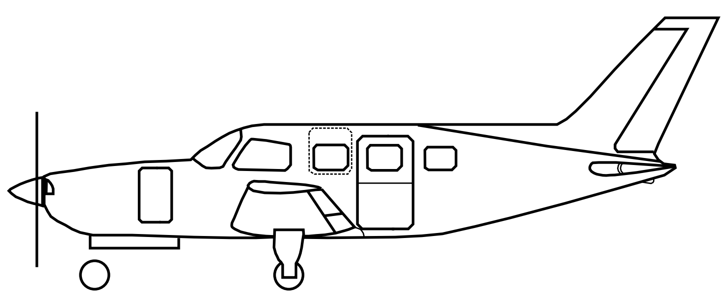
GTX-330 - Single Transponder
Figure 2 (Sheet 2 of 3)



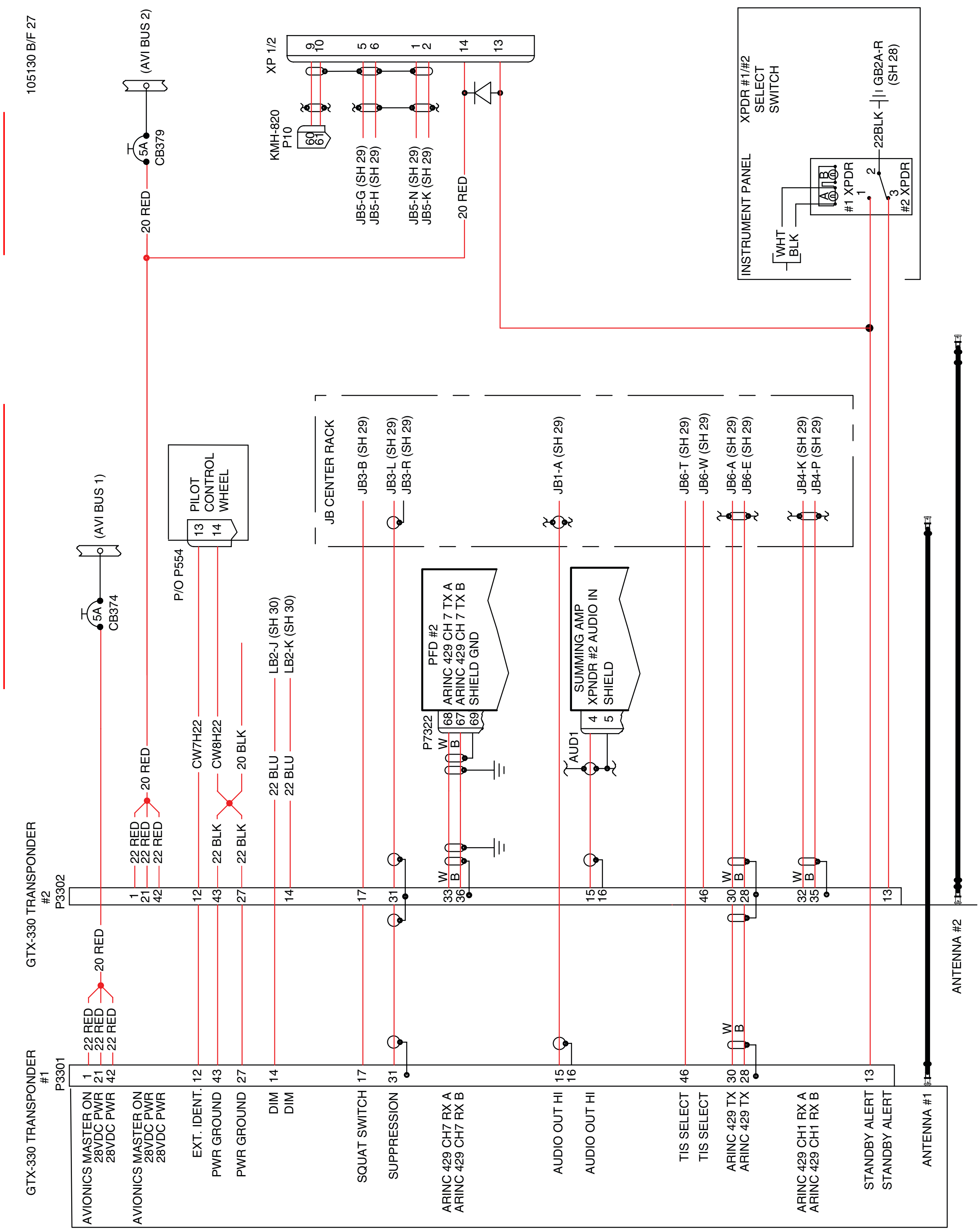
GTX-330 - Single Transponder
Figure 2 (Sheet 3 of 3)

Item #	Designation	Description

Information Pending - See Parts Catalog



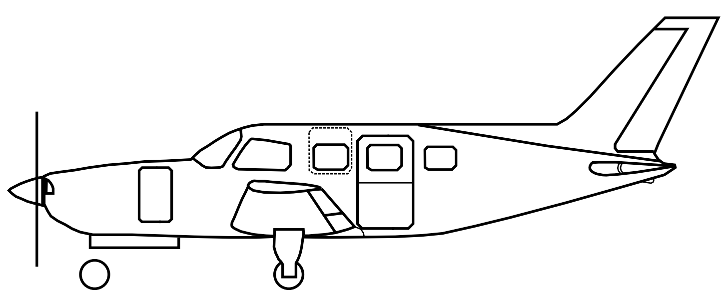
GTX-330 - Dual Transponder
Figure 3 (Sheet 1 of 2)



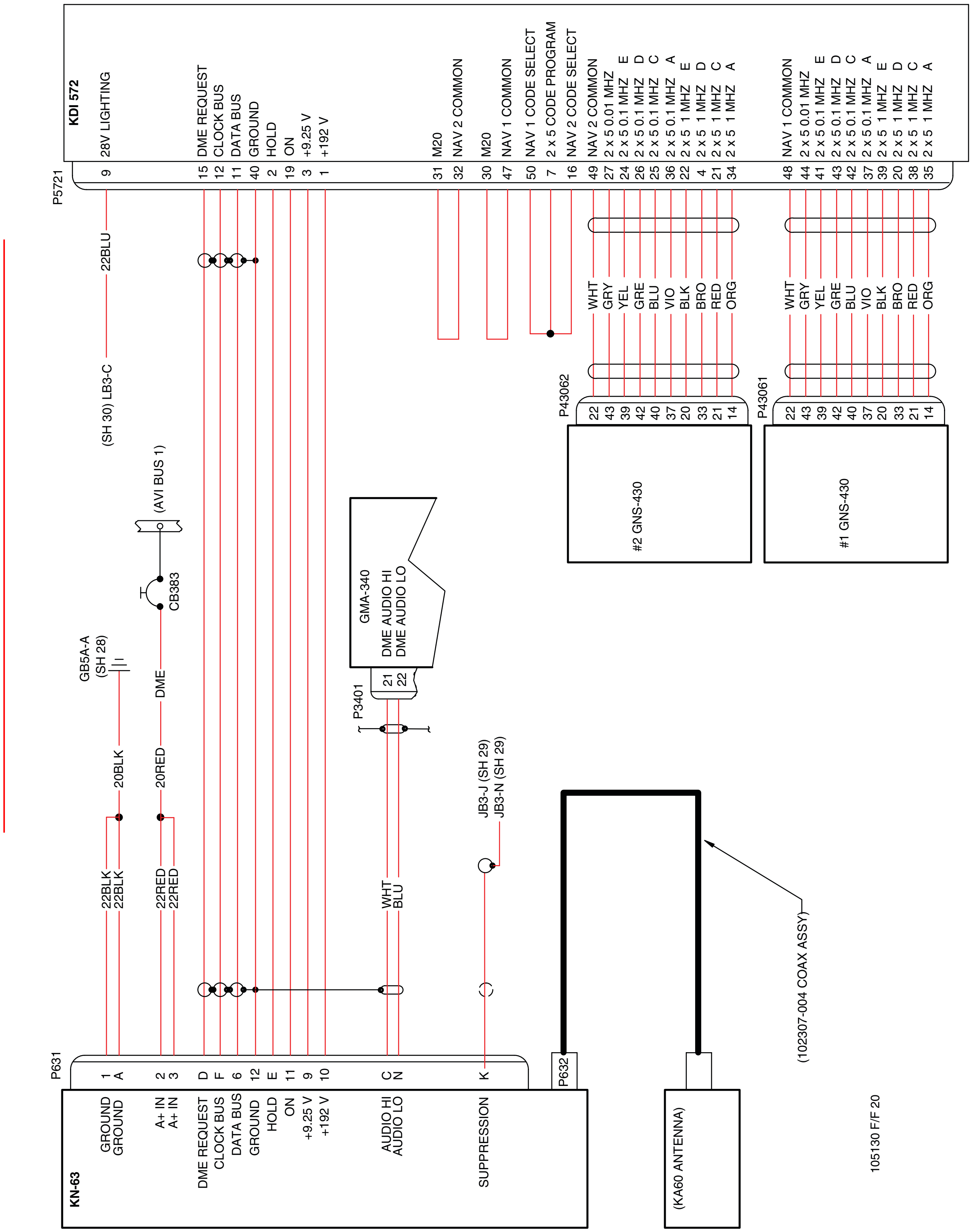
GTX-330 - Dual Transponder
Figure 3 (Sheet 2 of 2)

Item #	Designation	Description

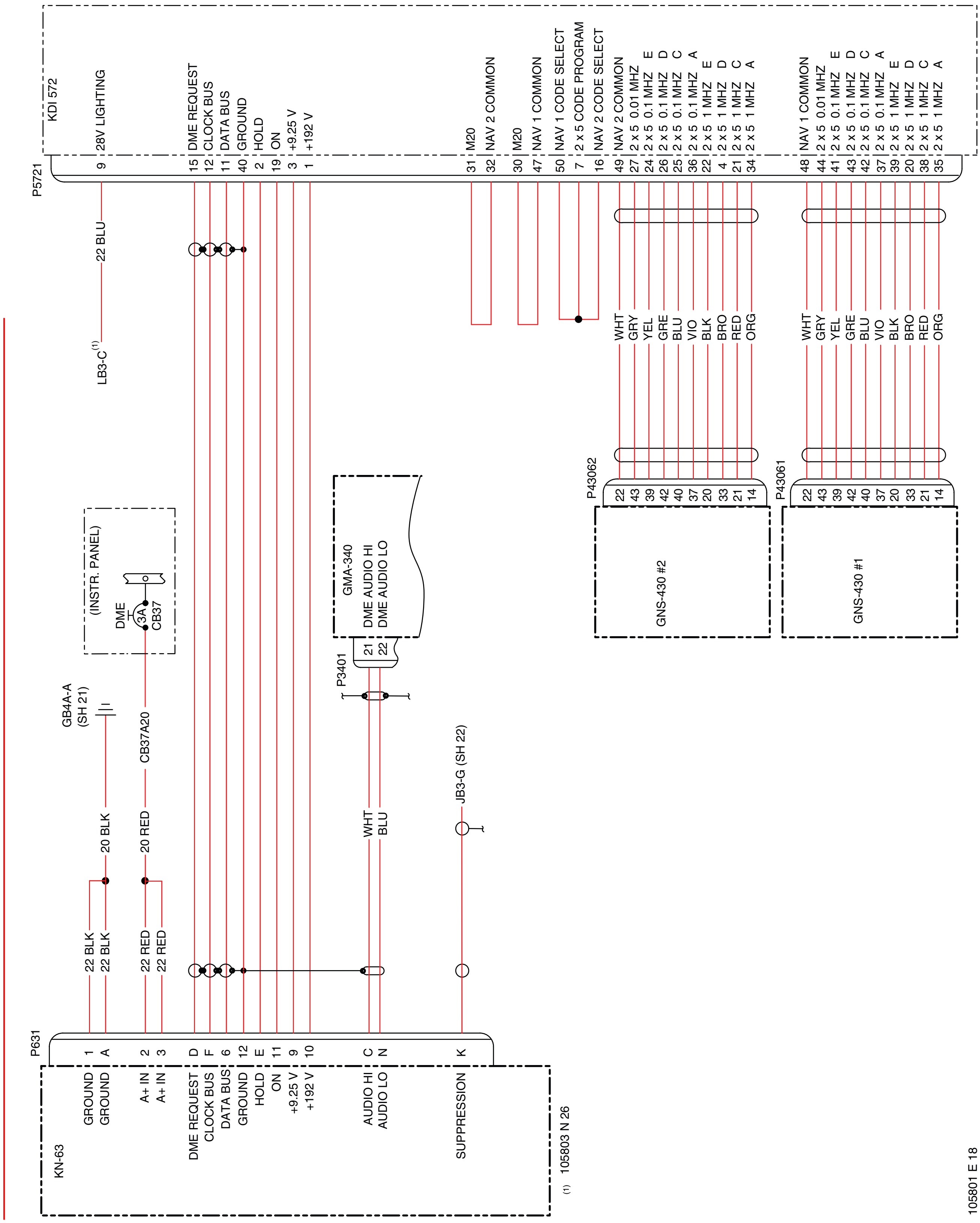
Information Pending - See Parts Catalog



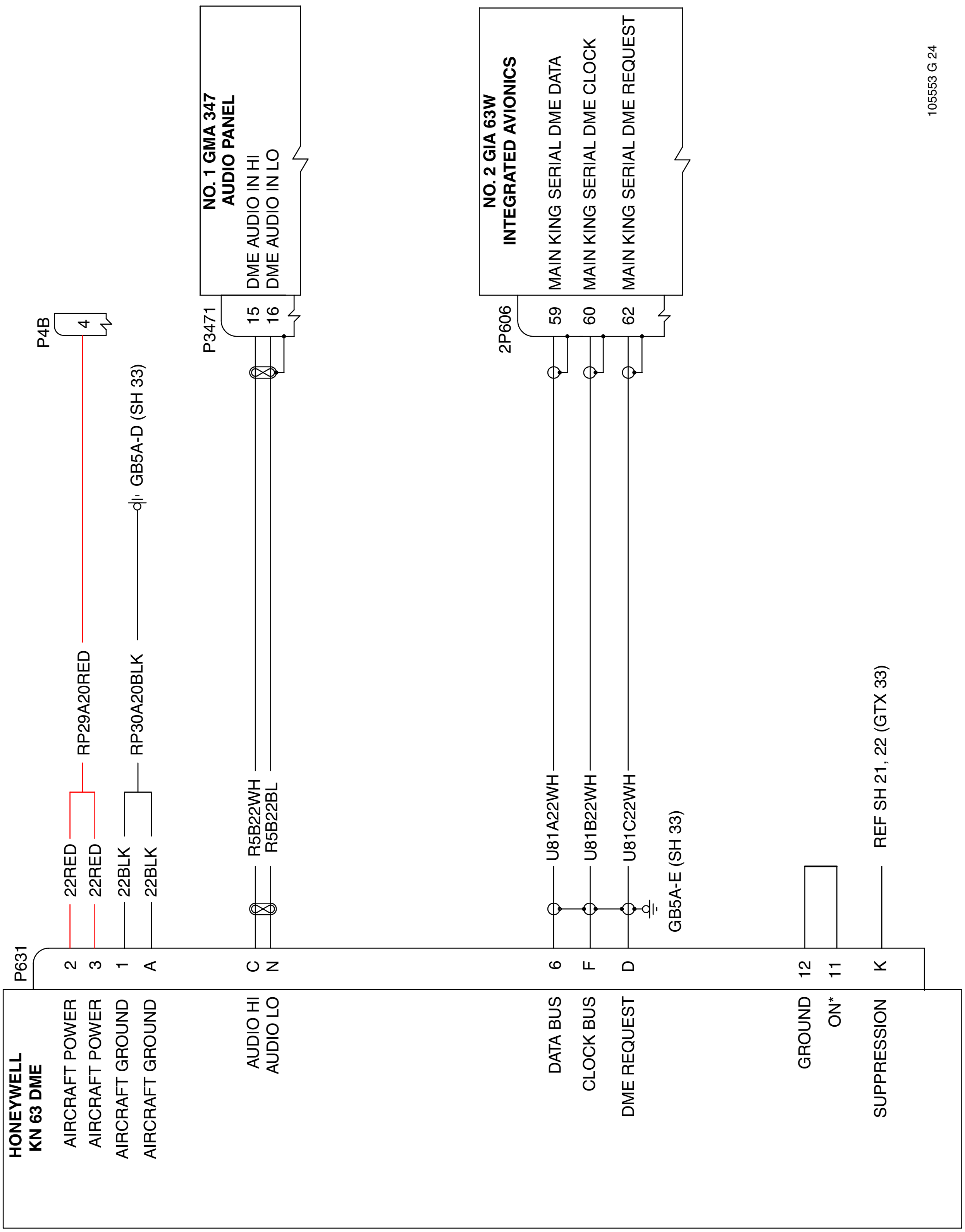
KN-63 - DME
Figure 4 (Sheet 1 of 5)



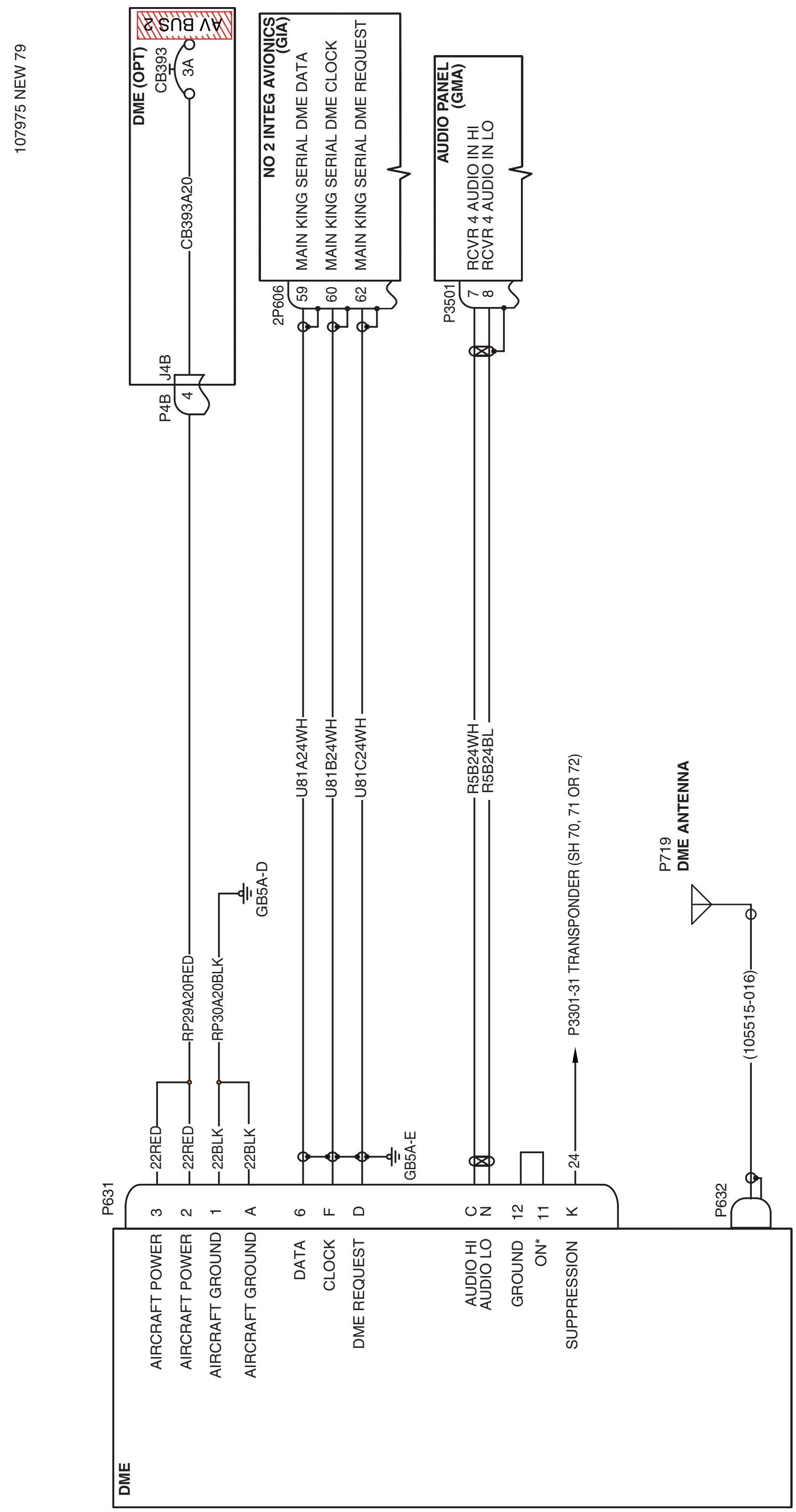
KN-63 - DME
Figure 4 (Sheet 2 of 5)



KN-63 - DME
Figure 4 (Sheet 3 of 5)



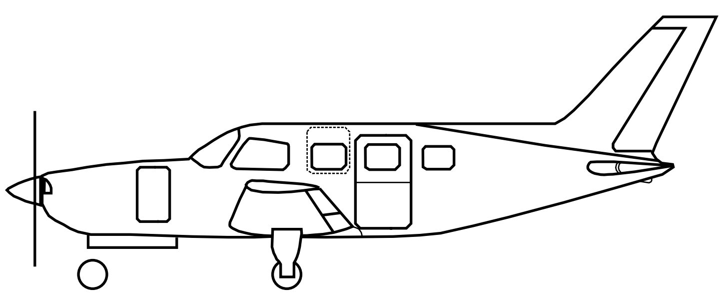
KN-63 - DME
Figure 4 (Sheet 4 of 5)



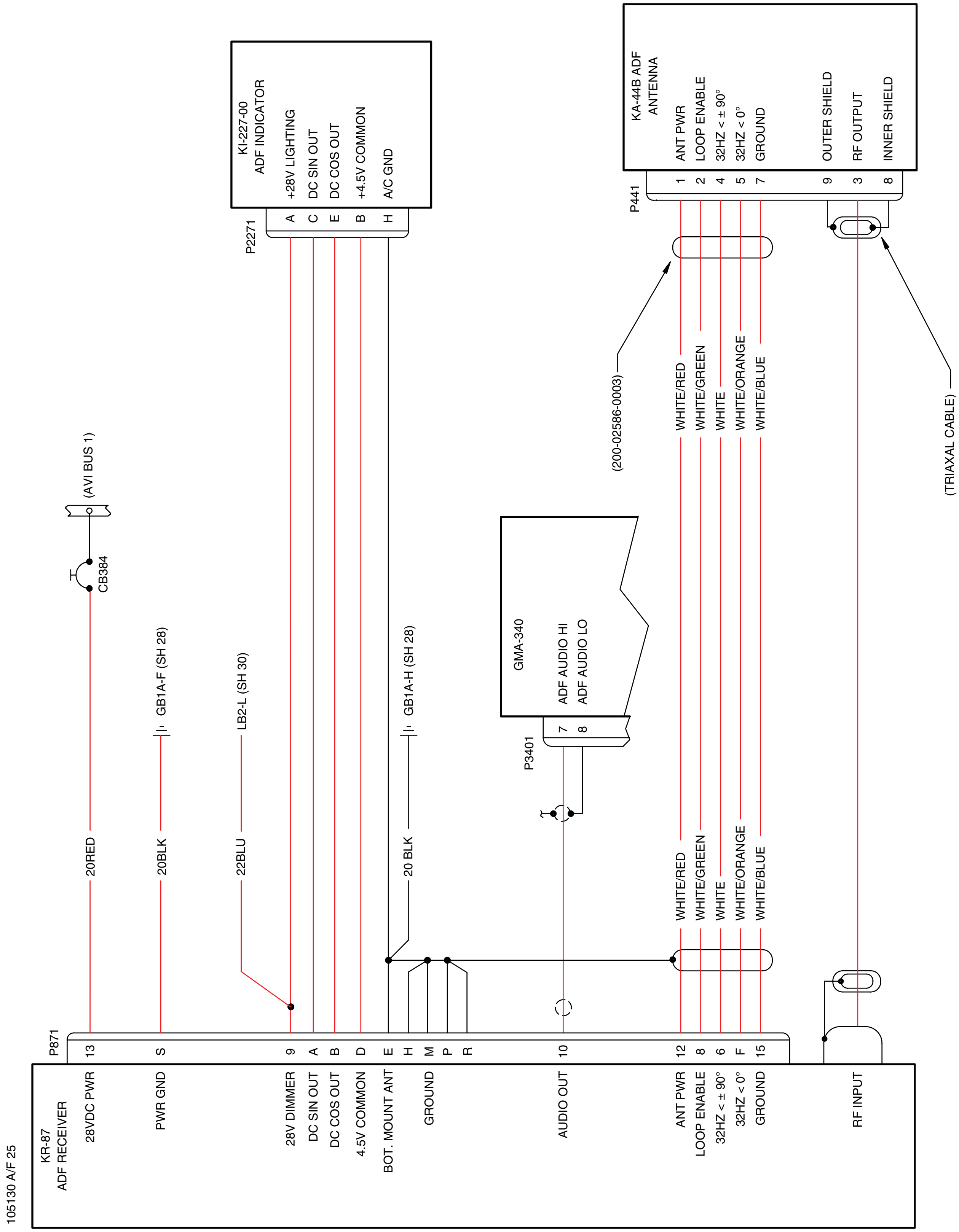
KN-63 - DME
Figure 4 (Sheet 5 of 5)

Item #	Designation	Description

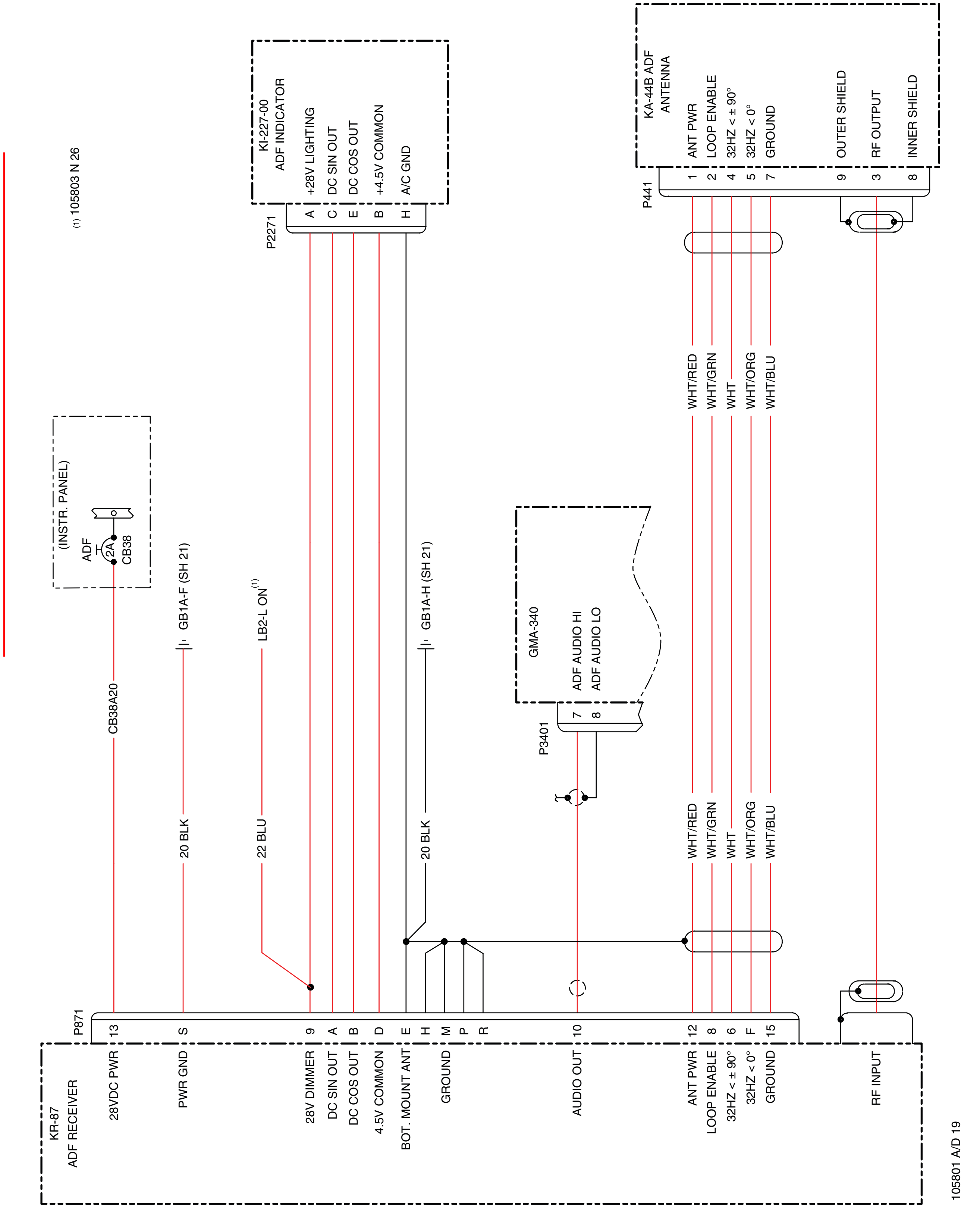
Information Pending - See Parts Catalog



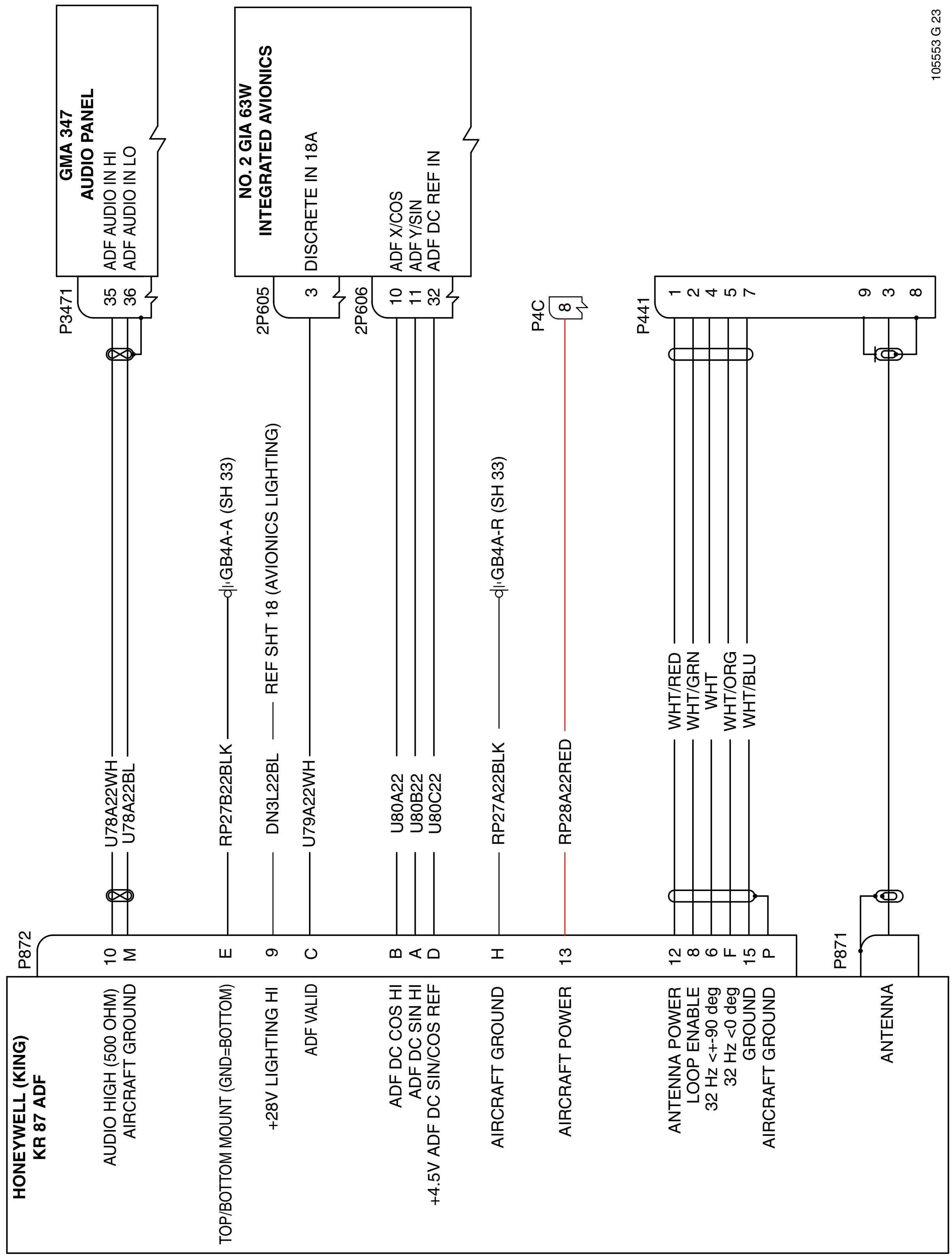
KR-87 - ADF
Figure 5 (Sheet 1 of 4)



KR-87 - ADF
Figure 5 (Sheet 2 of 4)



KR-87 - ADF
Figure 5 (Sheet 3 of 4)



KR-87 - ADF
Figure 5 (Sheet 4 of 4)

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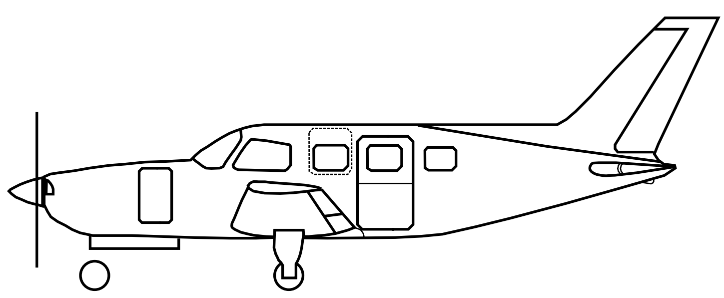
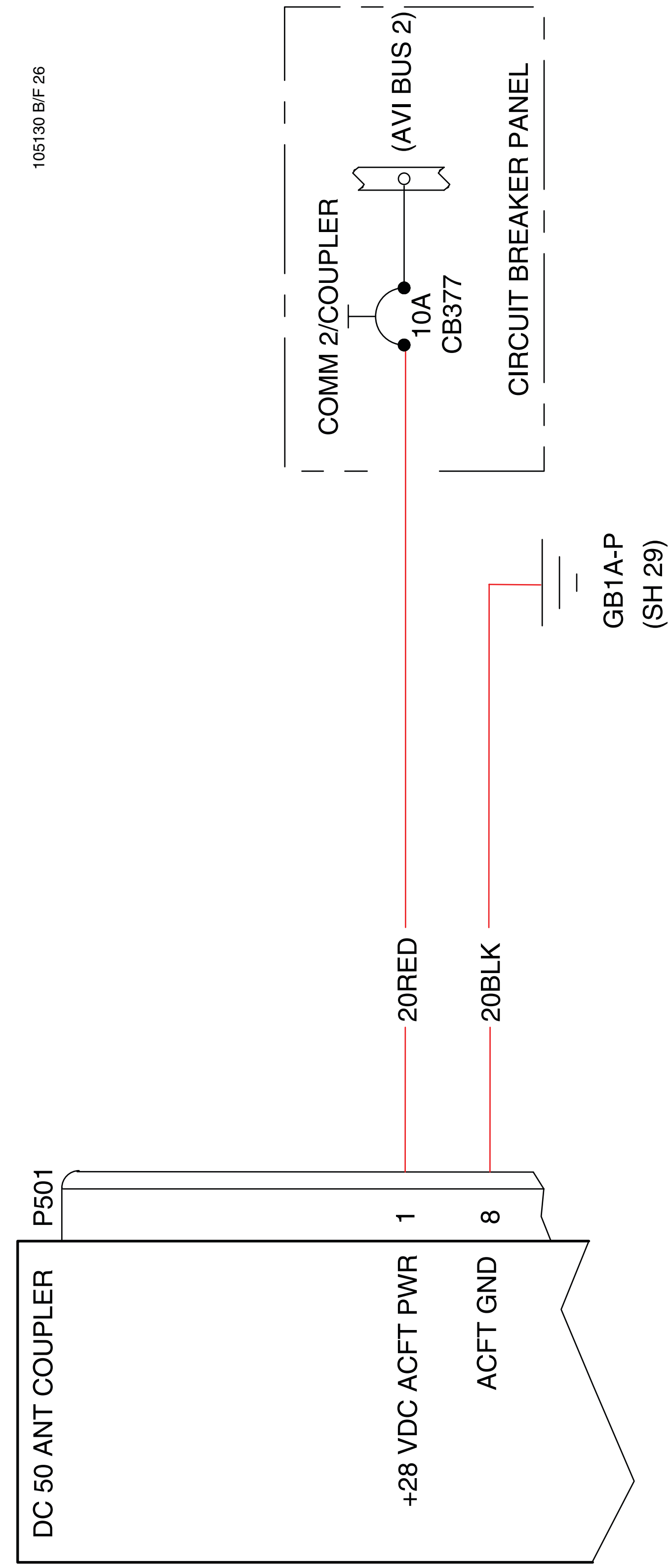
Item #	Designation	Description

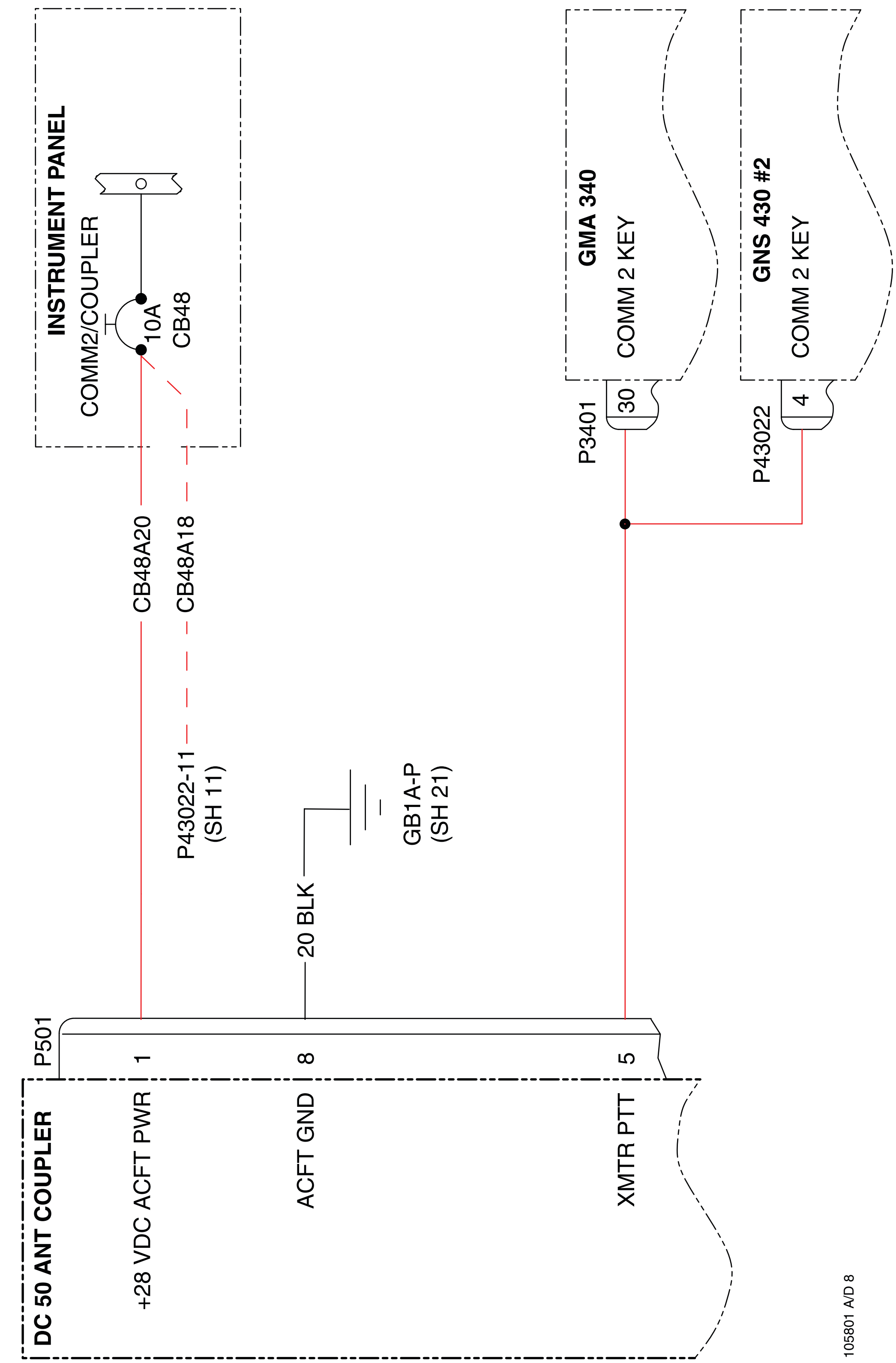
Information Pending - See Parts Catalog



DC Coupler
Figure 6 (Sheet 1 of 3)



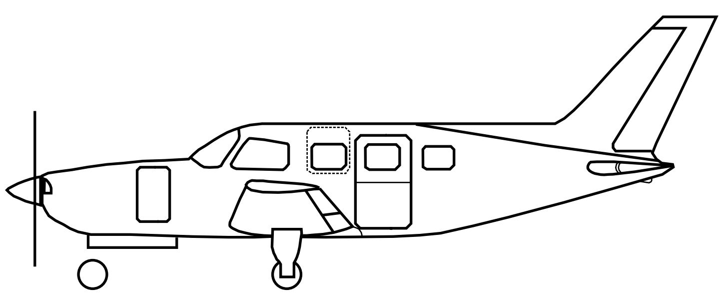
DC Coupler
Figure 6 (Sheet 2 of 3)



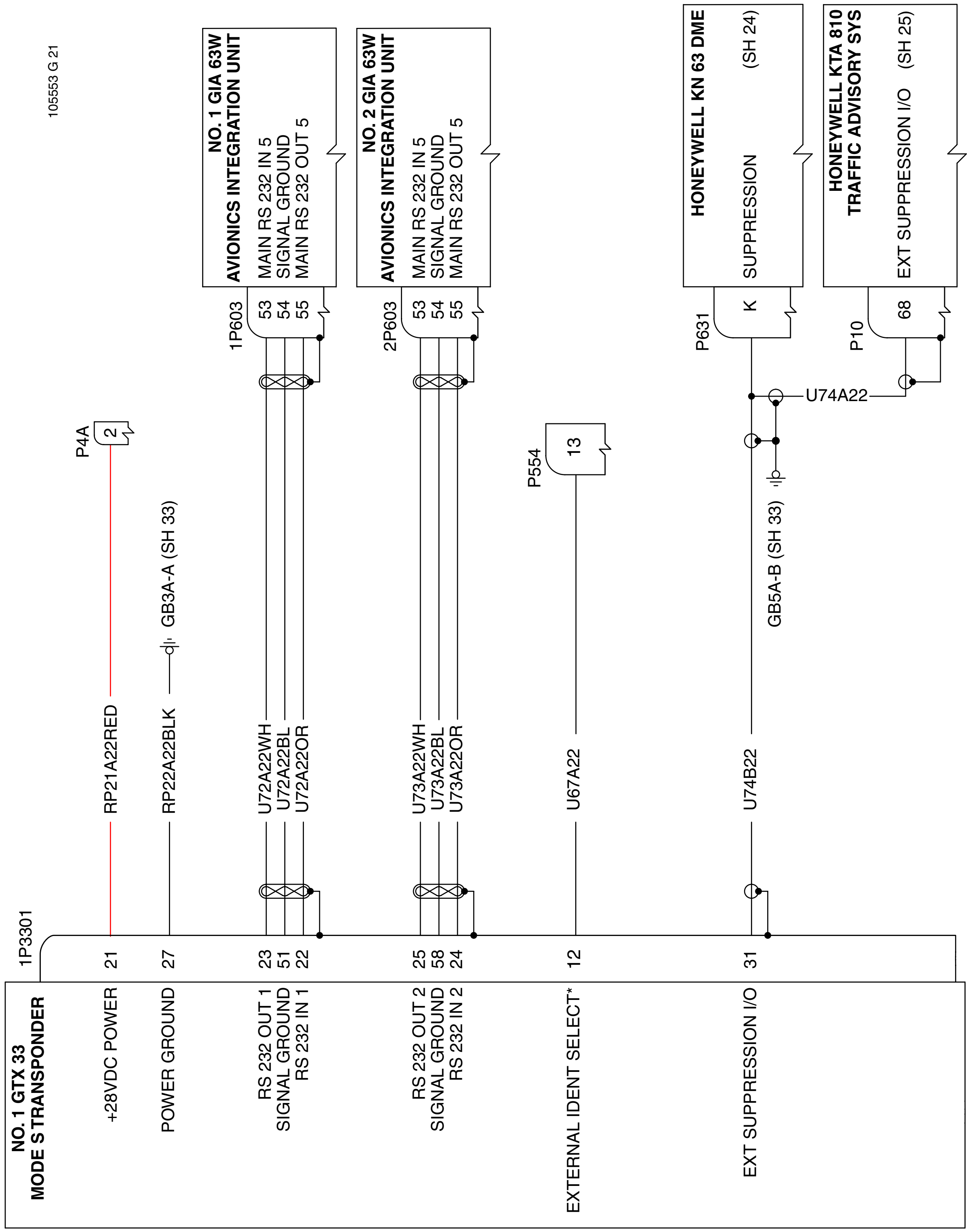
DC Coupler
Figure 6 (Sheet 3 of 3)

Item #	Designation	Description

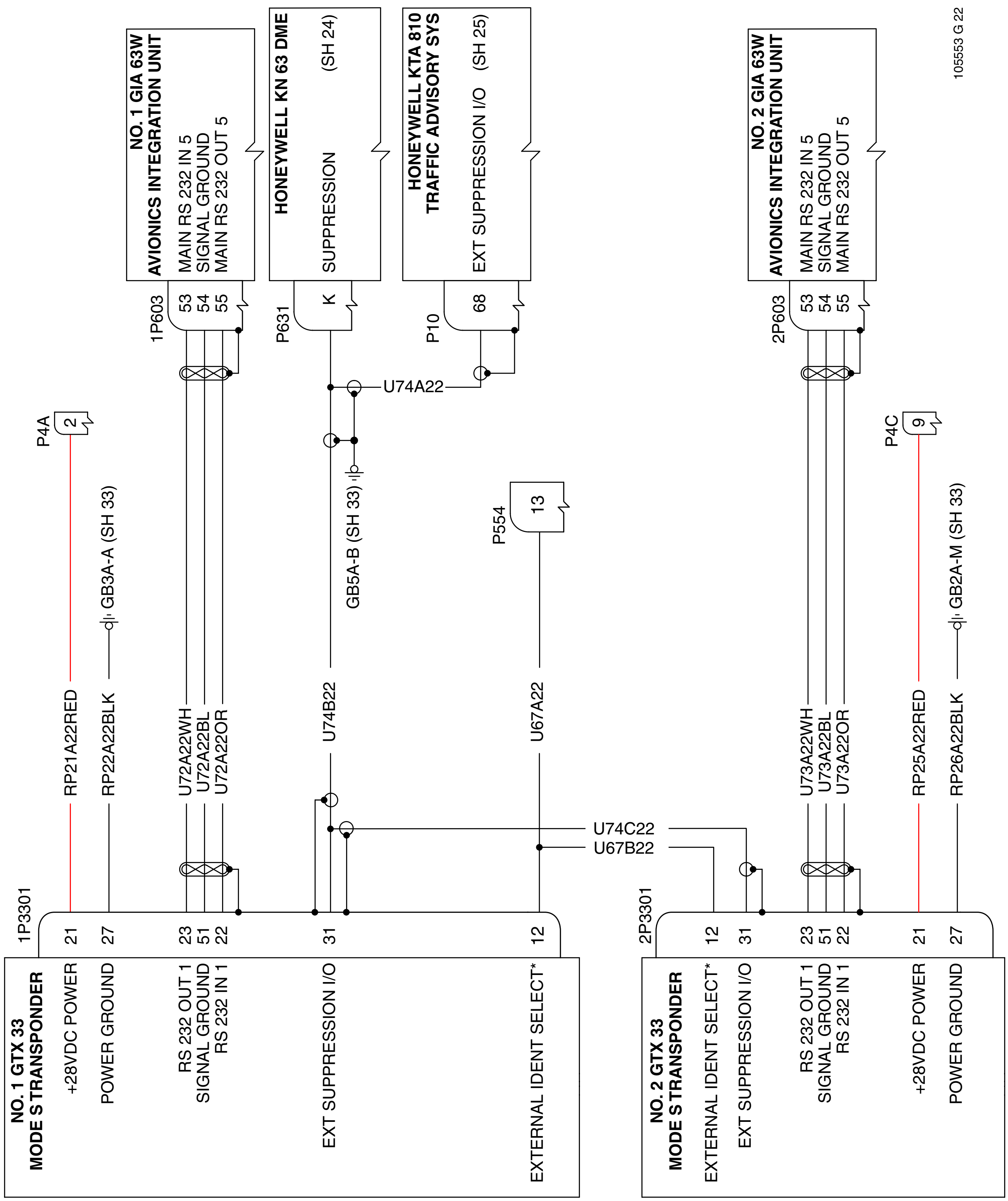
Information Pending - See Parts Catalog



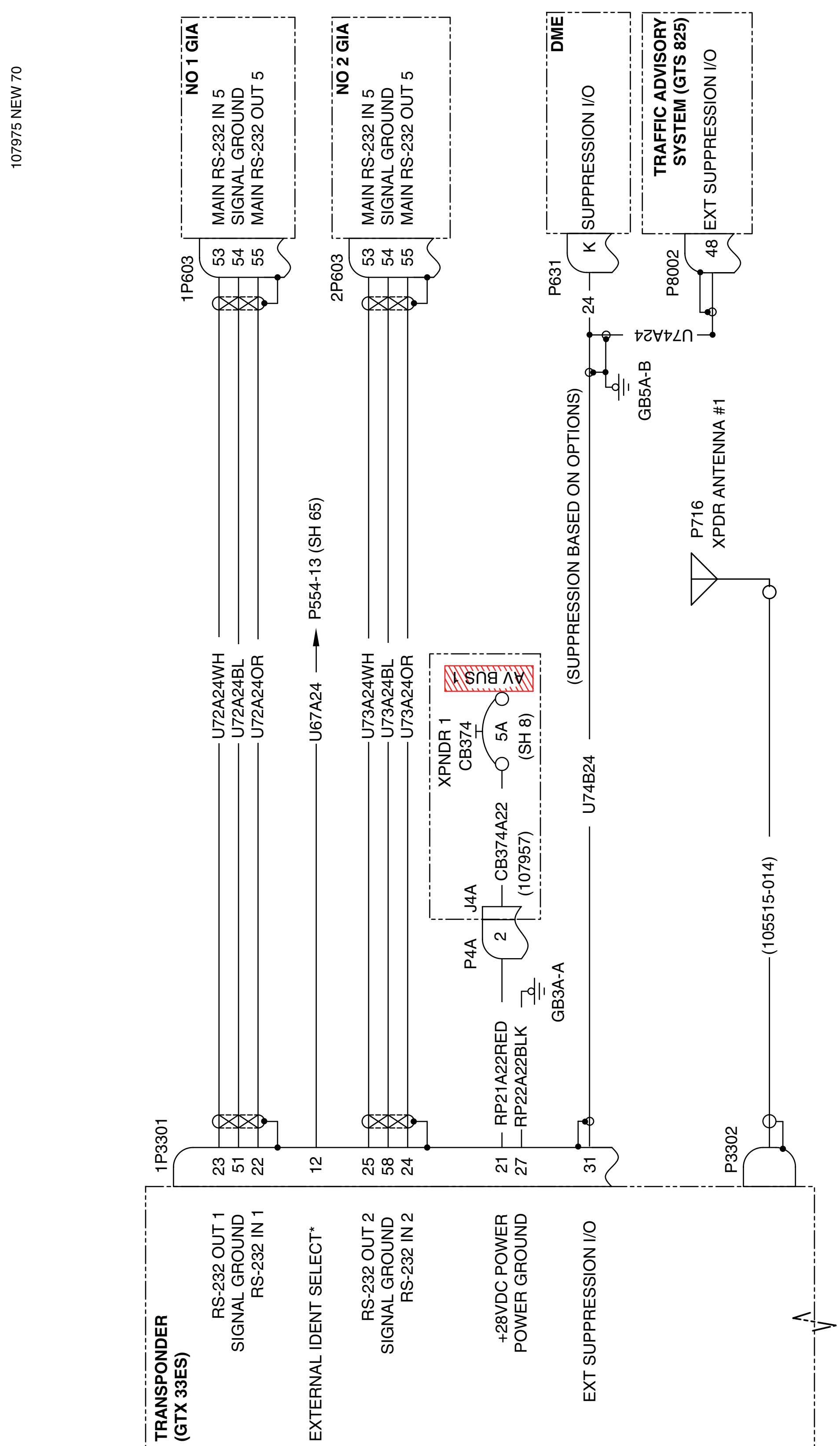
GTX-33 Transponder
Figure 7 (Sheet 1 of 6)



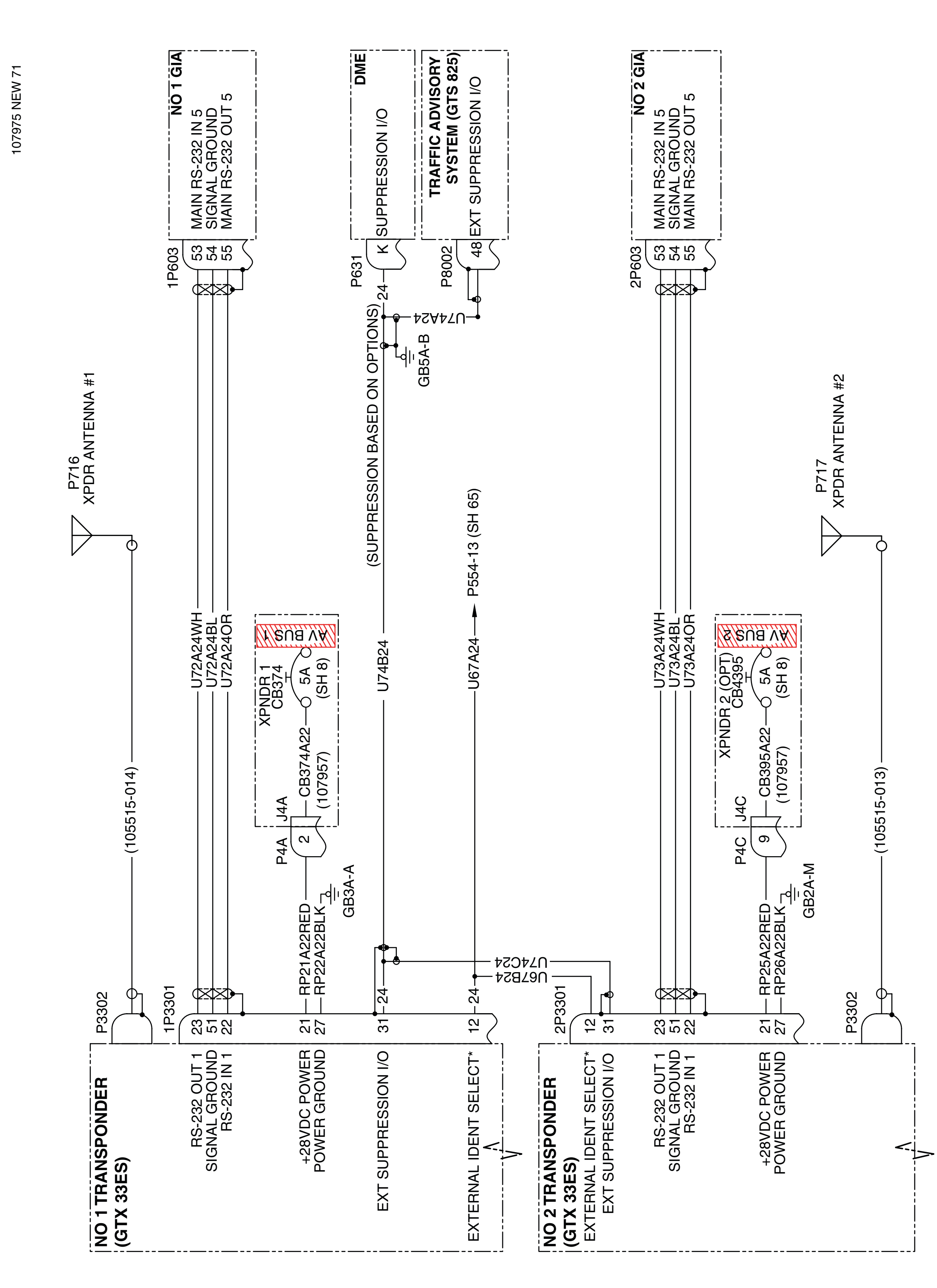
GTX-33 Transponder
Figure 7 (Sheet 2 of 6)



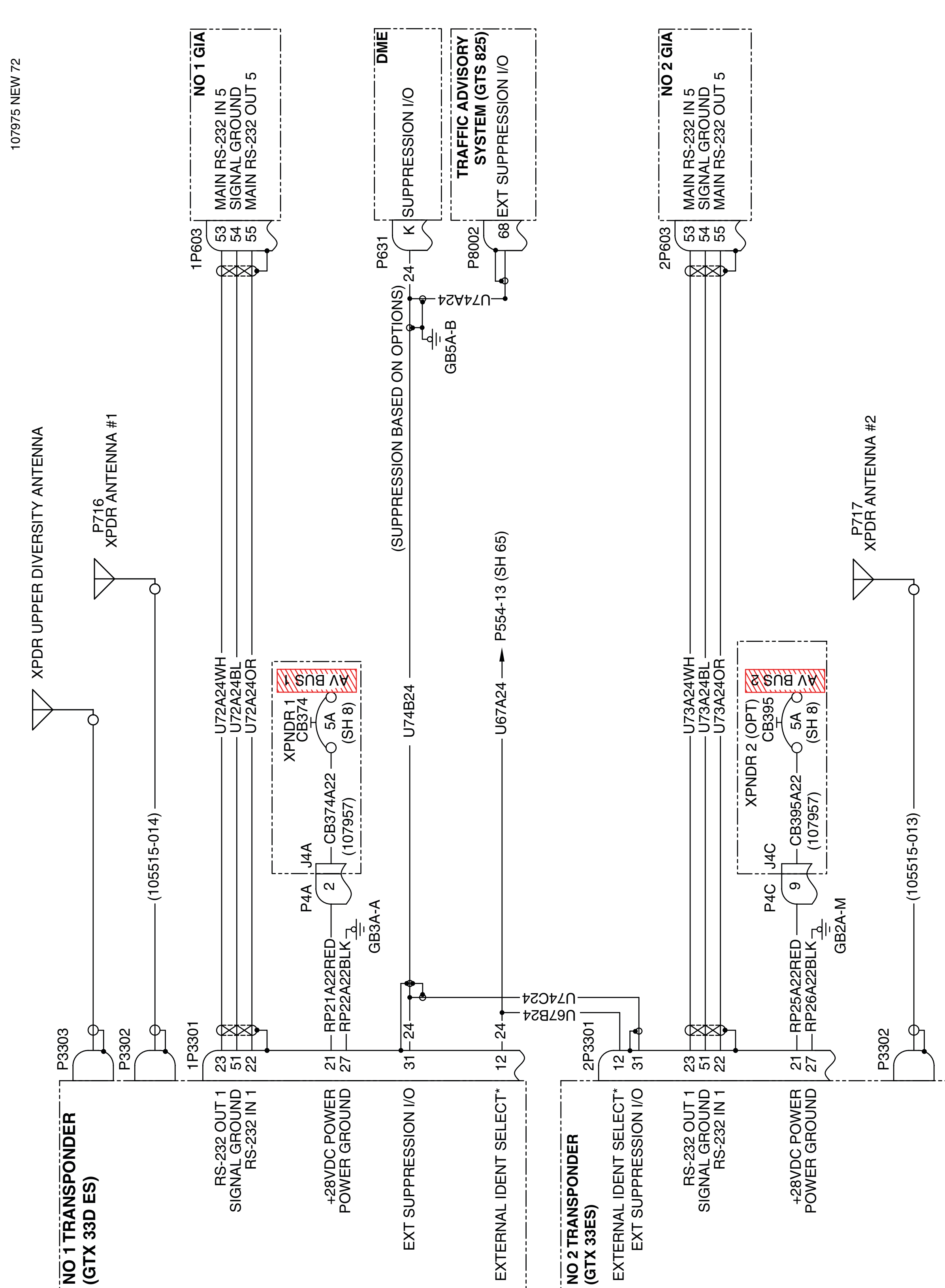
GTX-33 Transponder
Figure 7 (Sheet 3 of 6)



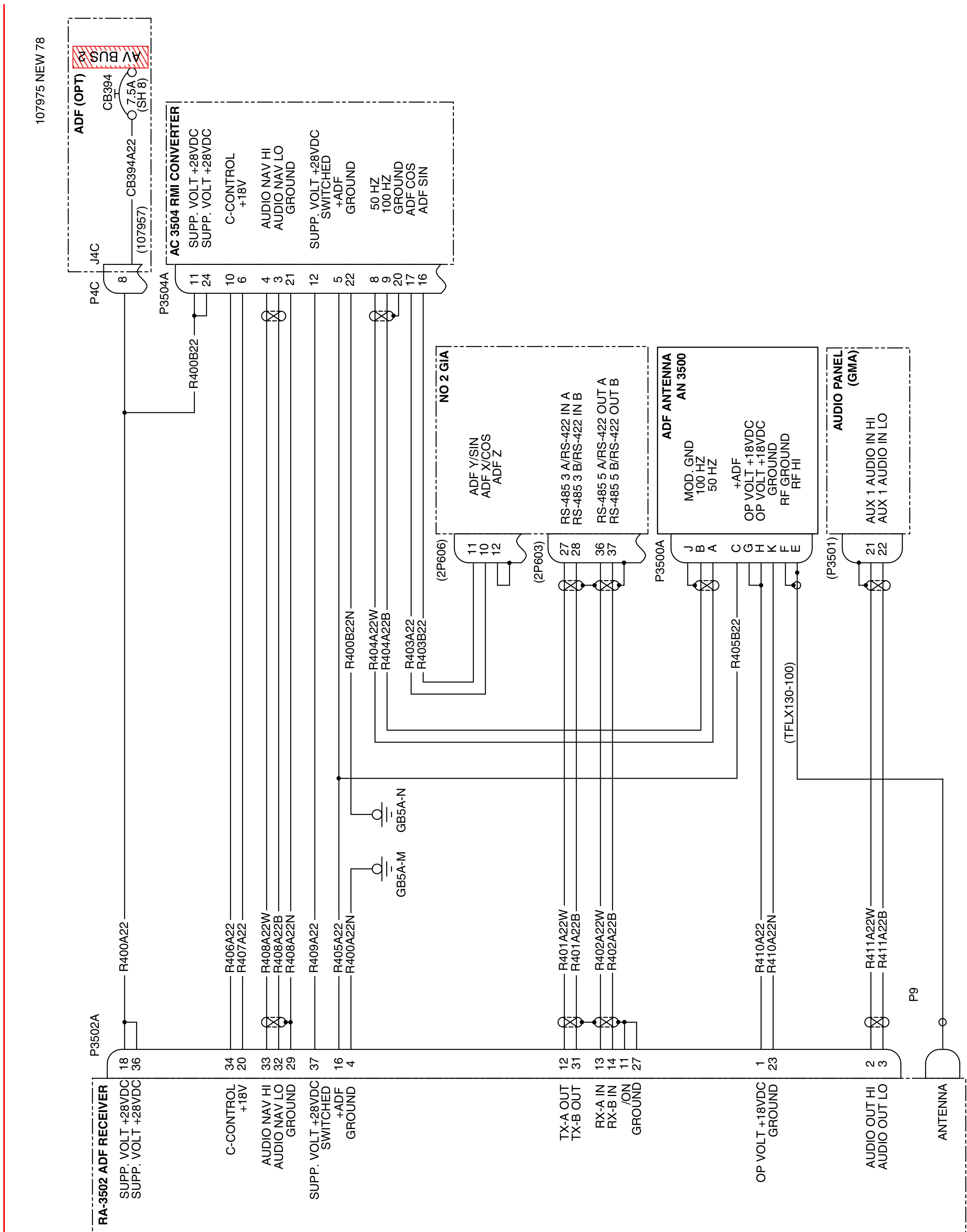
GTX-33 Transponder
Figure 7 (Sheet 4 of 6)



GTX-33 Transponder
Figure 7 (Sheet 5 of 6)



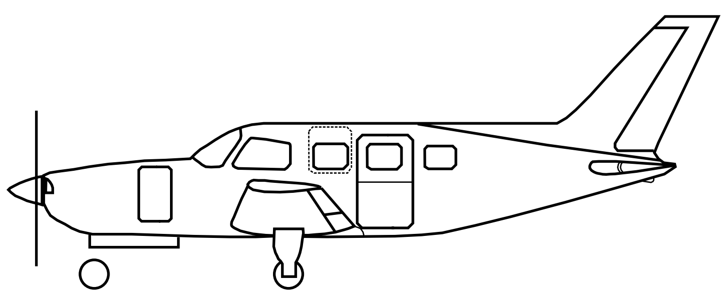
GTX-33 Transponder
Figure 7 (Sheet 6 of 6)



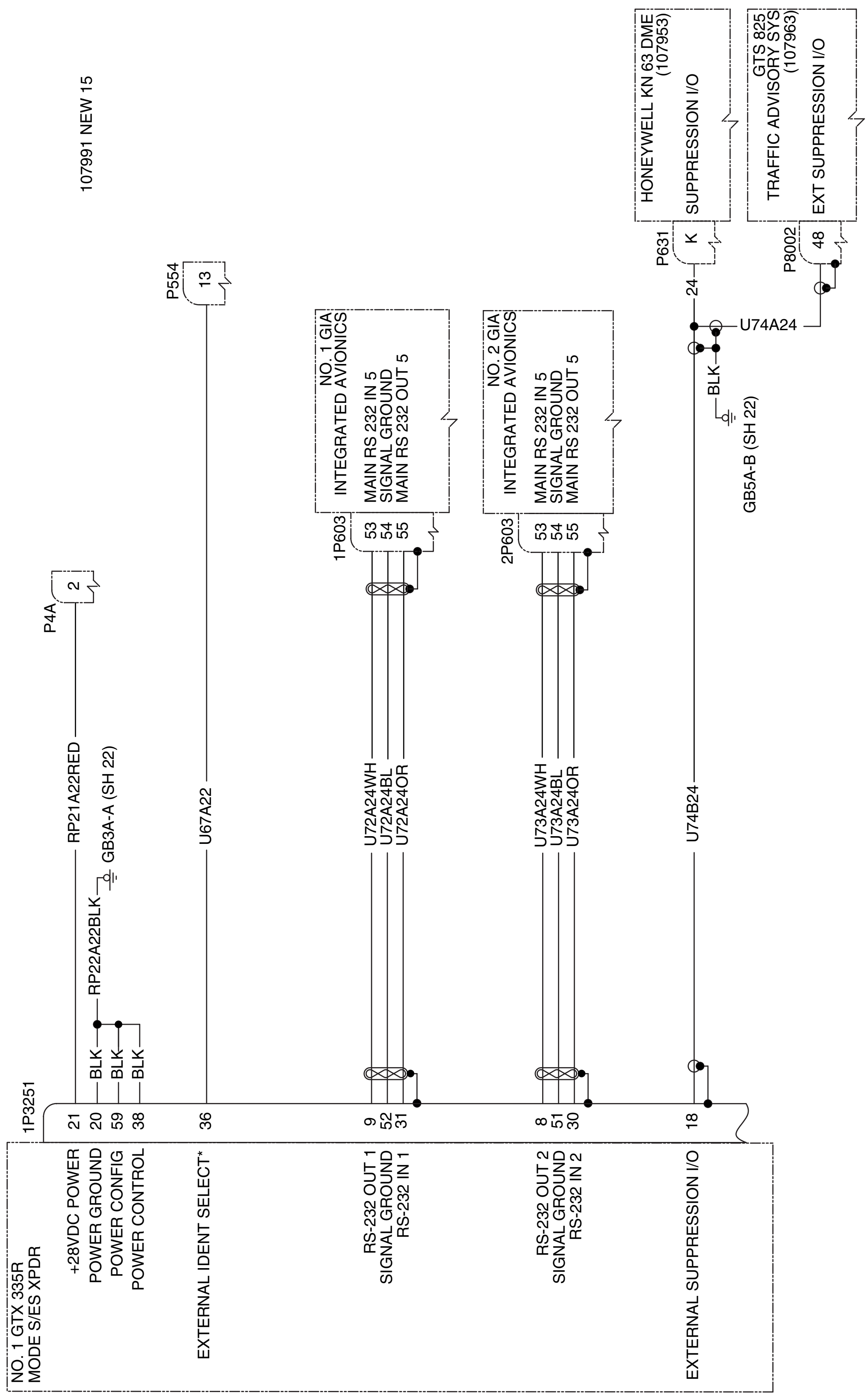
RA-3502 ADF (Optional)
Figure 8

Item #	Designation	Description

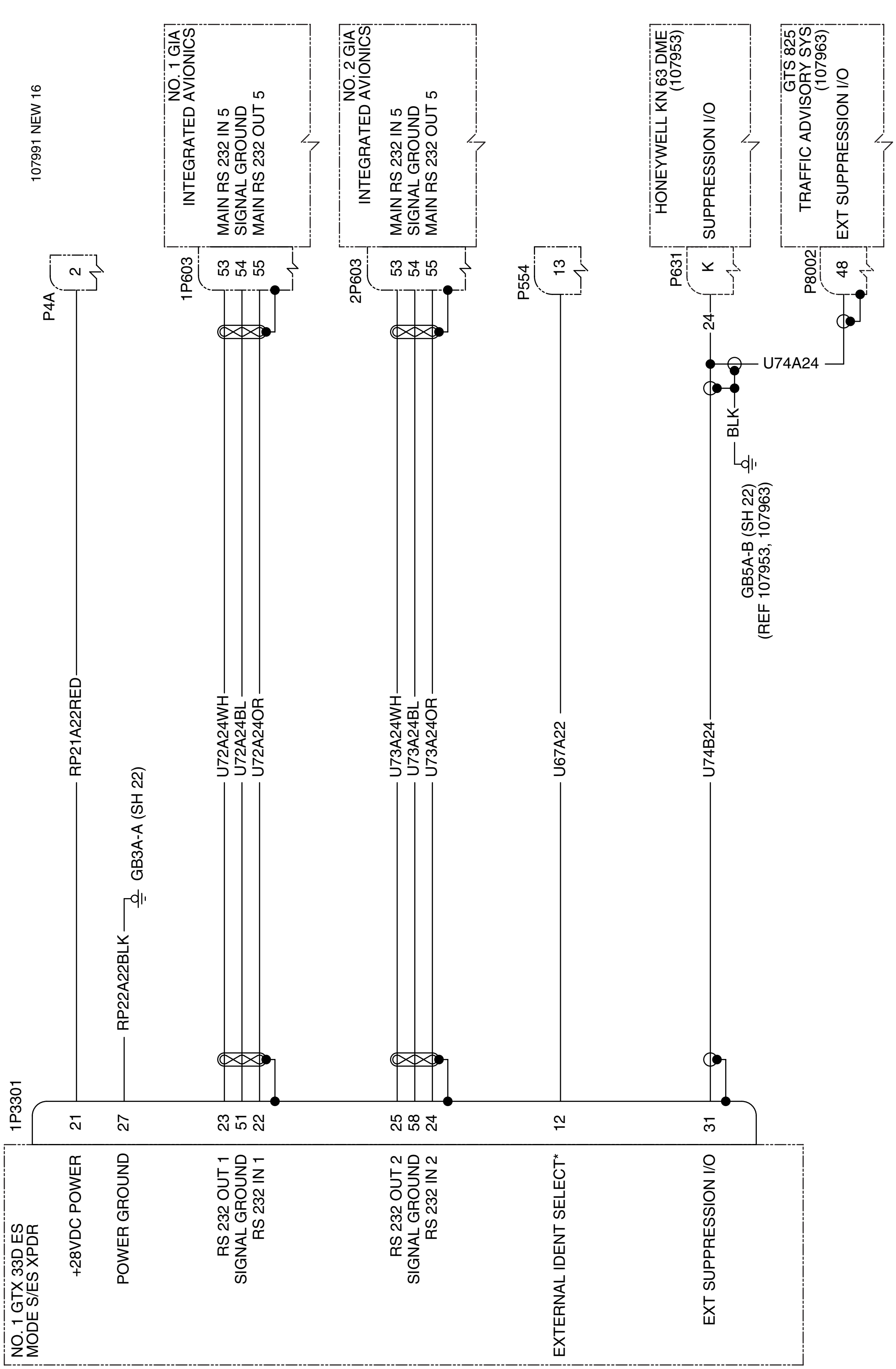
Information Pending - See Parts Catalog



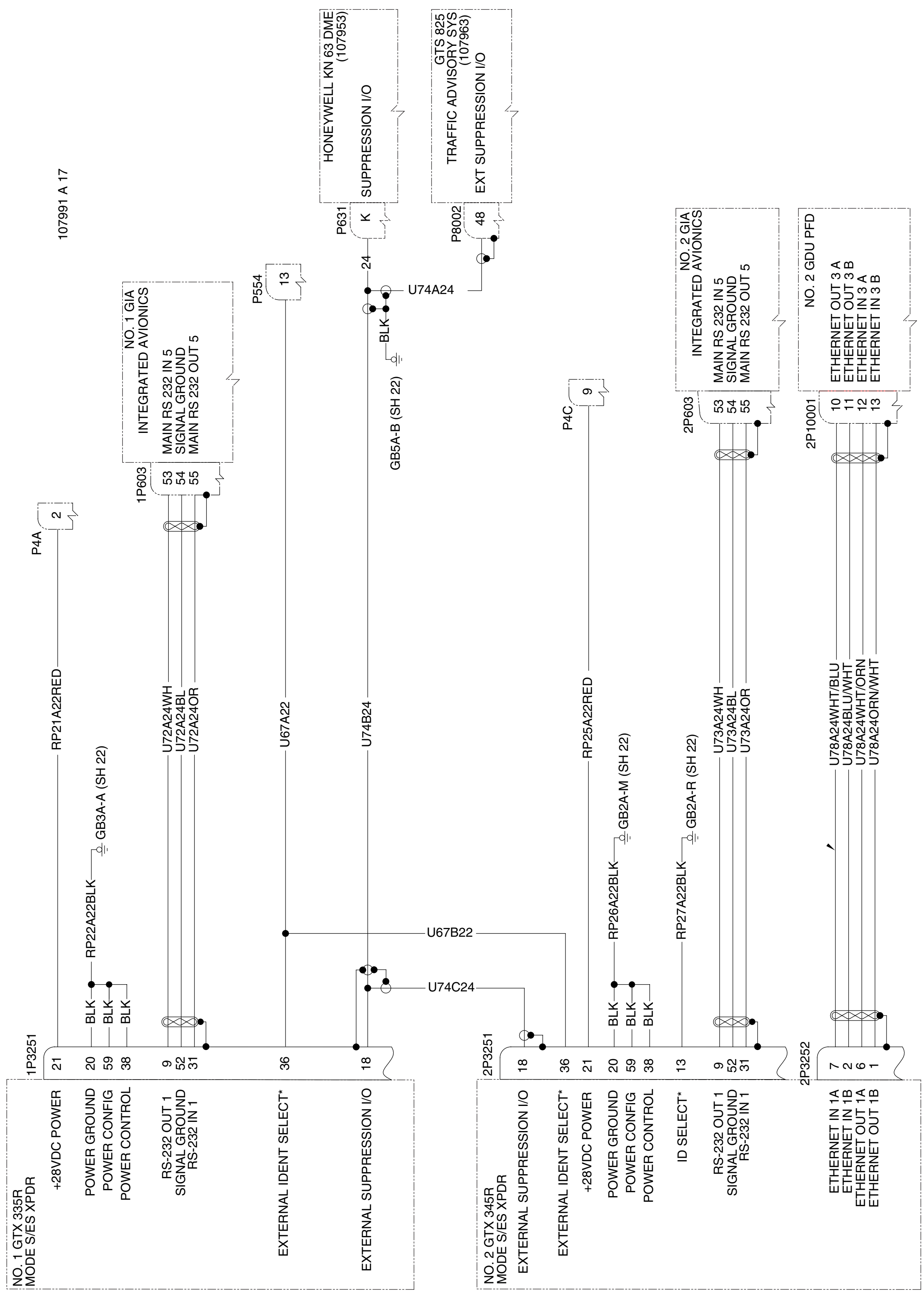
Transducer Options
Figure 9 (Sheet 1 of 5)



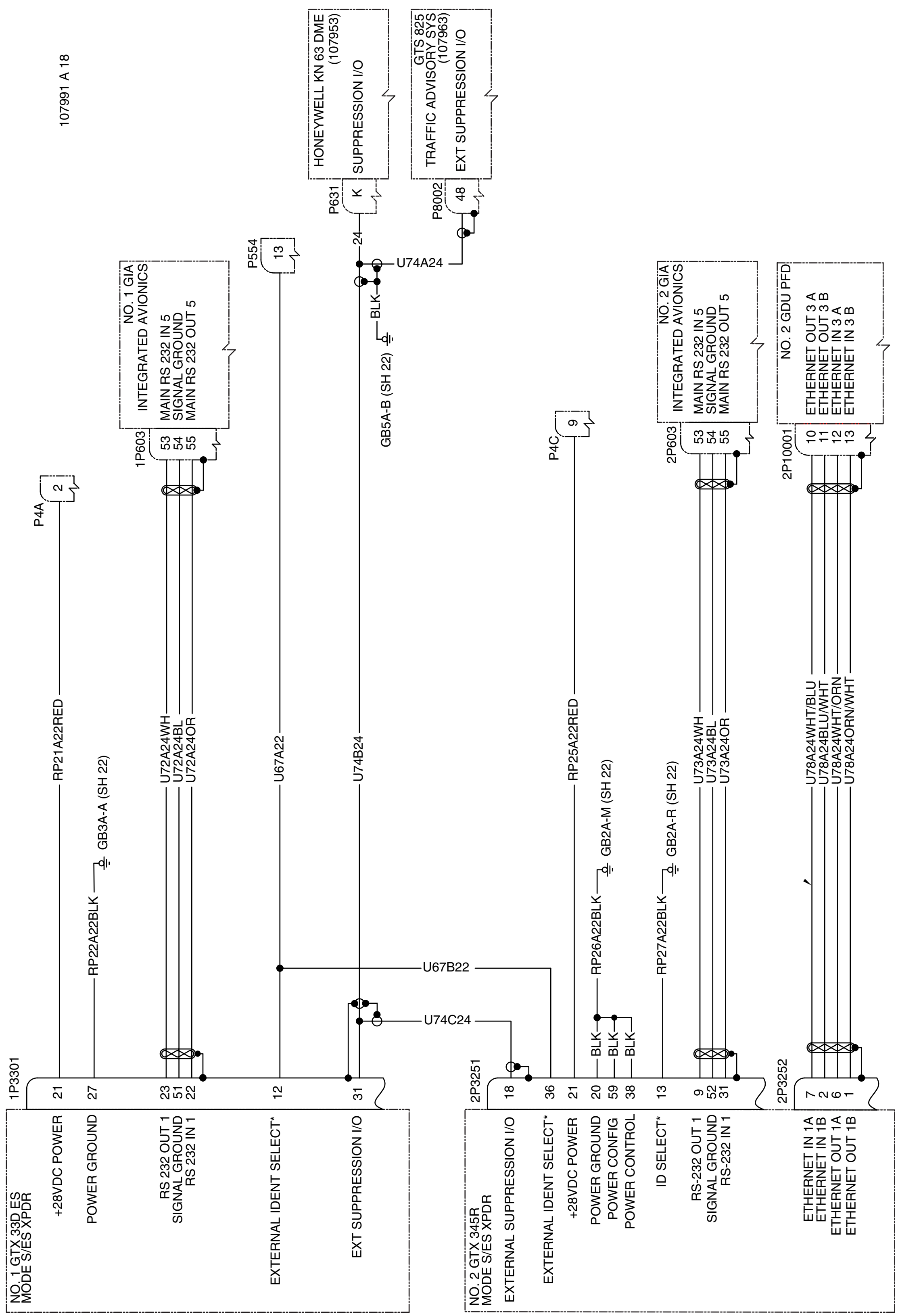
Transducer Options
Figure 9 (Sheet 2 of 5)



Transducer Options
Figure 9 (Sheet 3 of 5)



Transducer Options
Figure 9 (Sheet 4 of 5)



Transducer Options
Figure 9 (Sheet 5 of 5)

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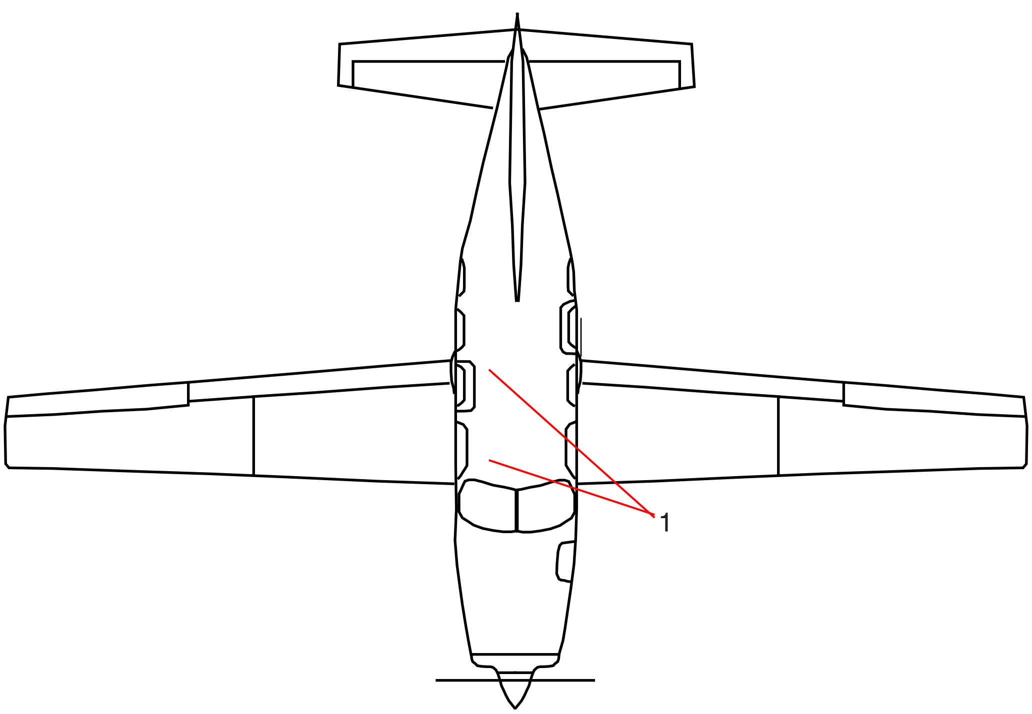
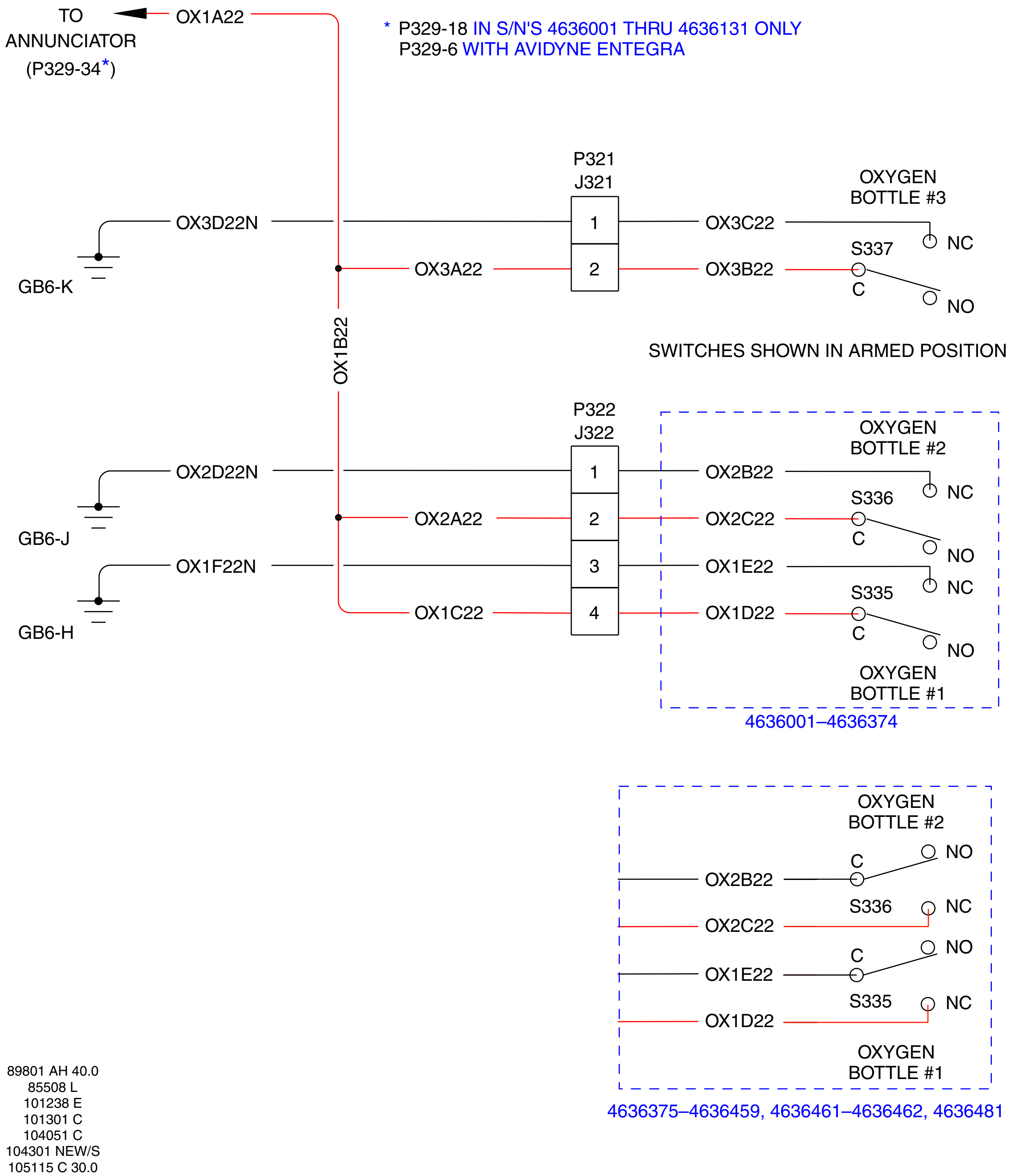
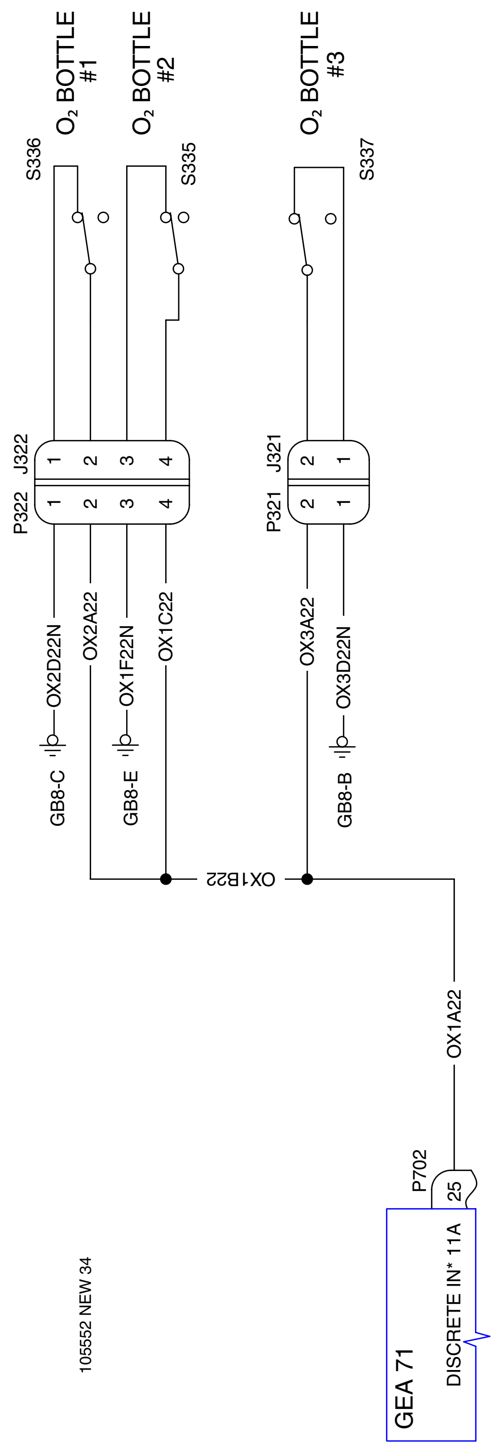
Item #	Designation	Description
1	S335, S336, S337	Switch (3 each)



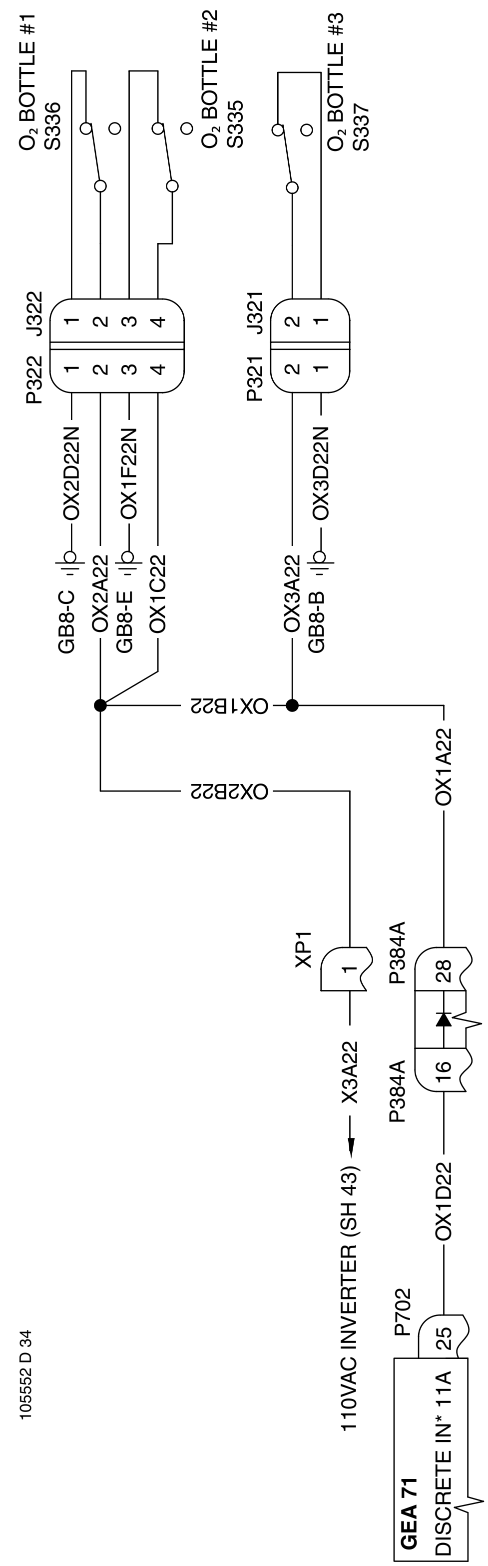
Oxygen System (Optional)
Figure 1 (Sheet 1 of 6)



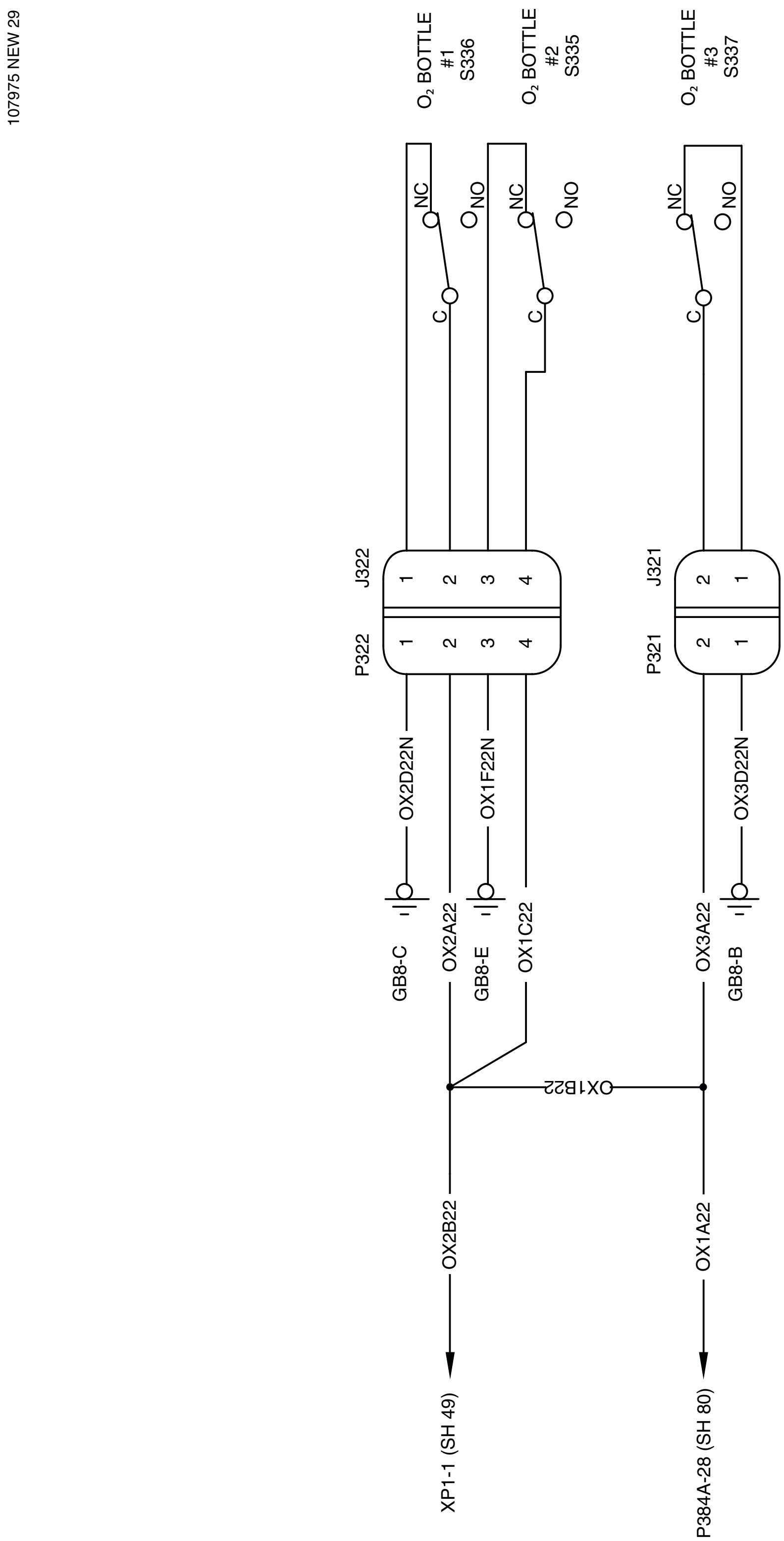
Oxygen System (Optional)
Figure 1 (Sheet 2 of 6)



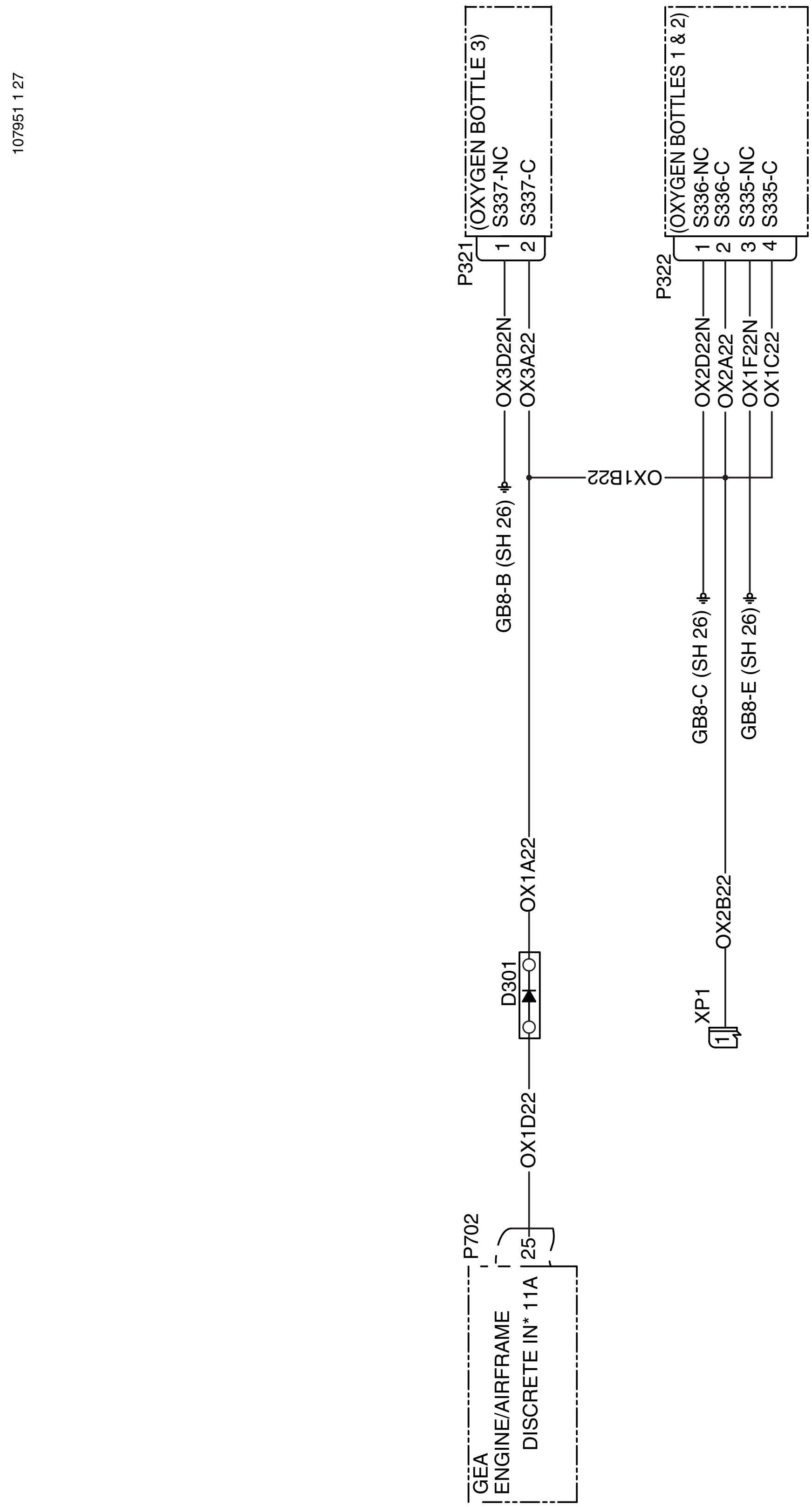
Oxygen System (Optional)
Figure 1 (Sheet 3 of 6)



Oxygen System (Optional)
Figure 1 (Sheet 4 of 6)

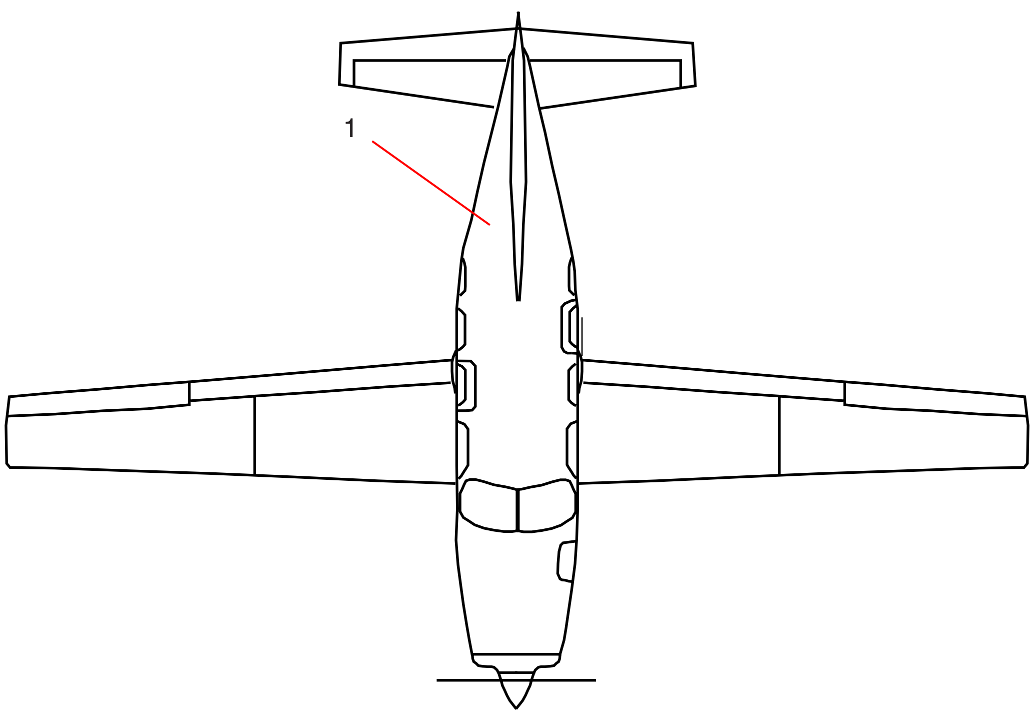


Oxygen System (Optional)
Figure 1 (Sheet 5 of 6)



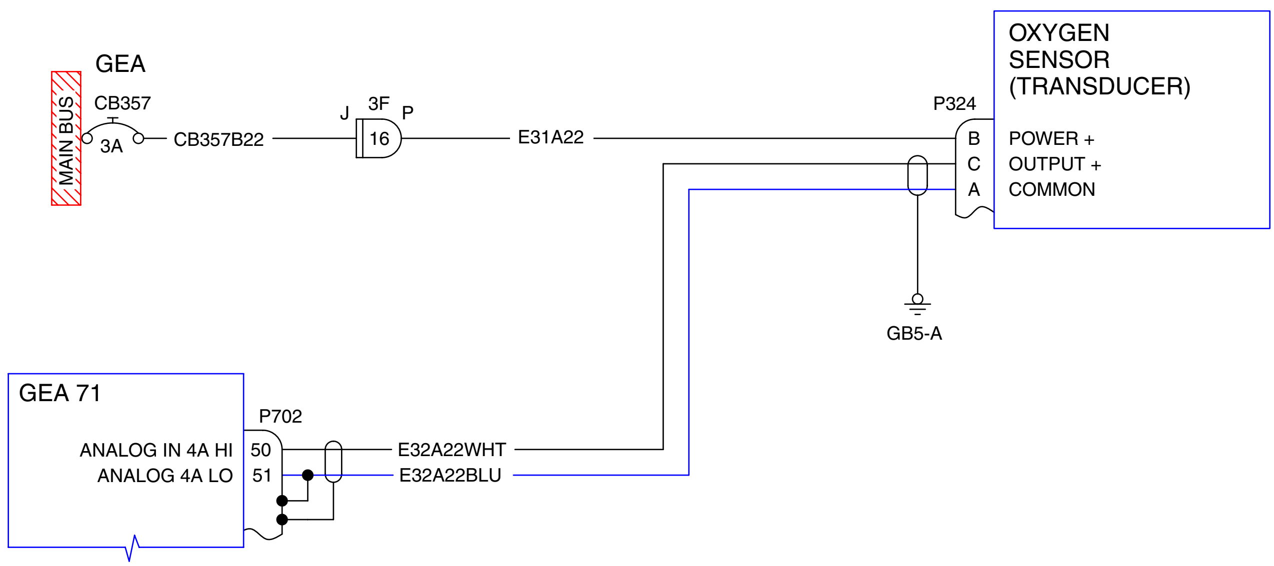
Oxygen System (Optional)
Figure 1 (Sheet 6 of 6)

Item #	Designation	Description
1	P324	Oxygen Sensor (Transducer)



Oxygen Pressure
Figure 1 (Sheet 1 of 2)

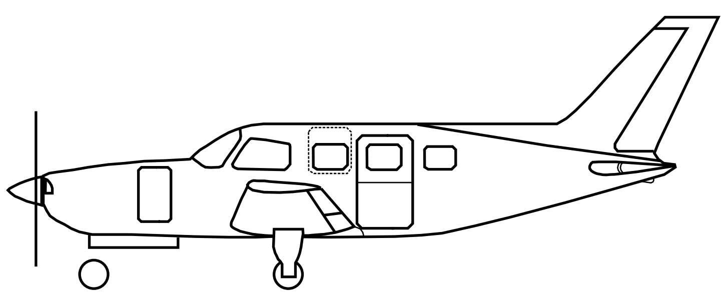
105552 NEW 25



Oxygen Pressure
Figure 1 (Sheet 2 of 2)

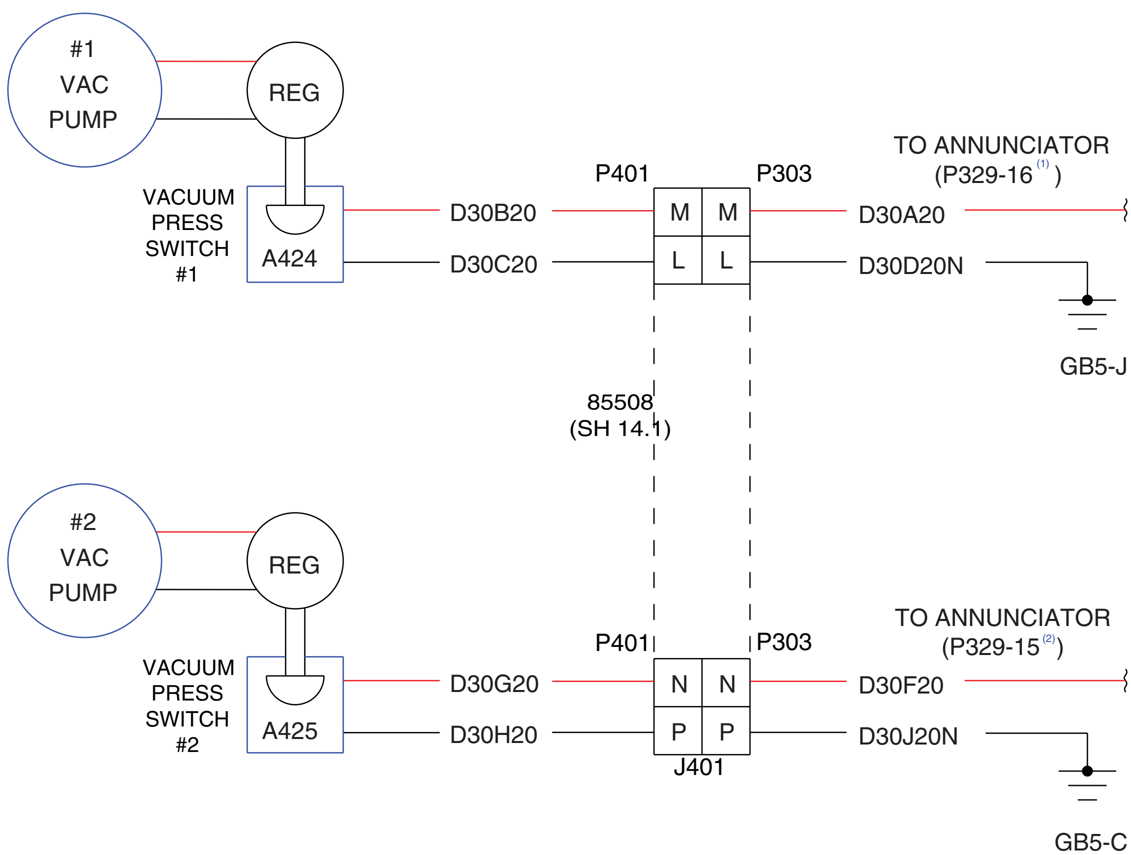
Item #	Designation	Description

Information Pending - See Parts Catalog



Vacuum Indication
Figure 1 (Sheet 1 of 4)

(1) P329-6 IN S/N'S 4636021 THRU 4636131 ONLY - 85508 L 27.0, 14.1
P328-16 PA46-350P WITH AVIDYNE ENTEGRA - 101238 E 27.0, 101301 C 27.0, 104051C 27.0
P328-11 PA46R-350T WITH AVIDYNE ENTEGRA - 105805 NEW 28.0

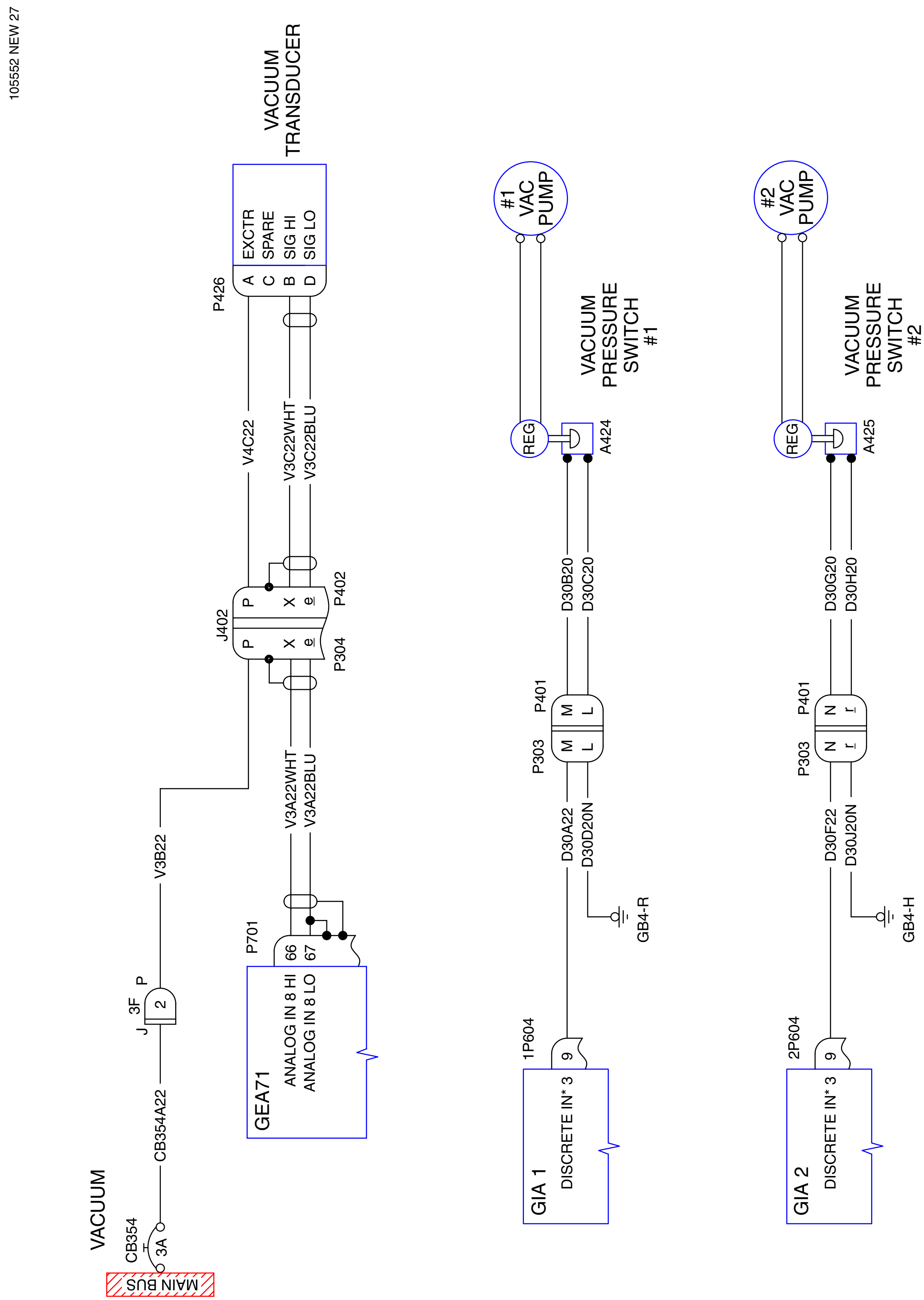


(2) P329-8 IN S/N'S 4636021 THRU 4636131 ONLY - 85508 L 45.0, 27.0, 14.1
P328-21 PA46-350P WITH AVIDYNE ENTEGRA - 101238 E 27.0, 101301 C 27.0, 104051C 27.0
P328-9 PA46R-350T WITH AVIDYNE ENTEGRA - 105805 NEW 28.0

85508 L 27.0
101238 E
101301 C
104051 C
104301 NEW/S
105115 A 24.0
105805 NEW/28.0

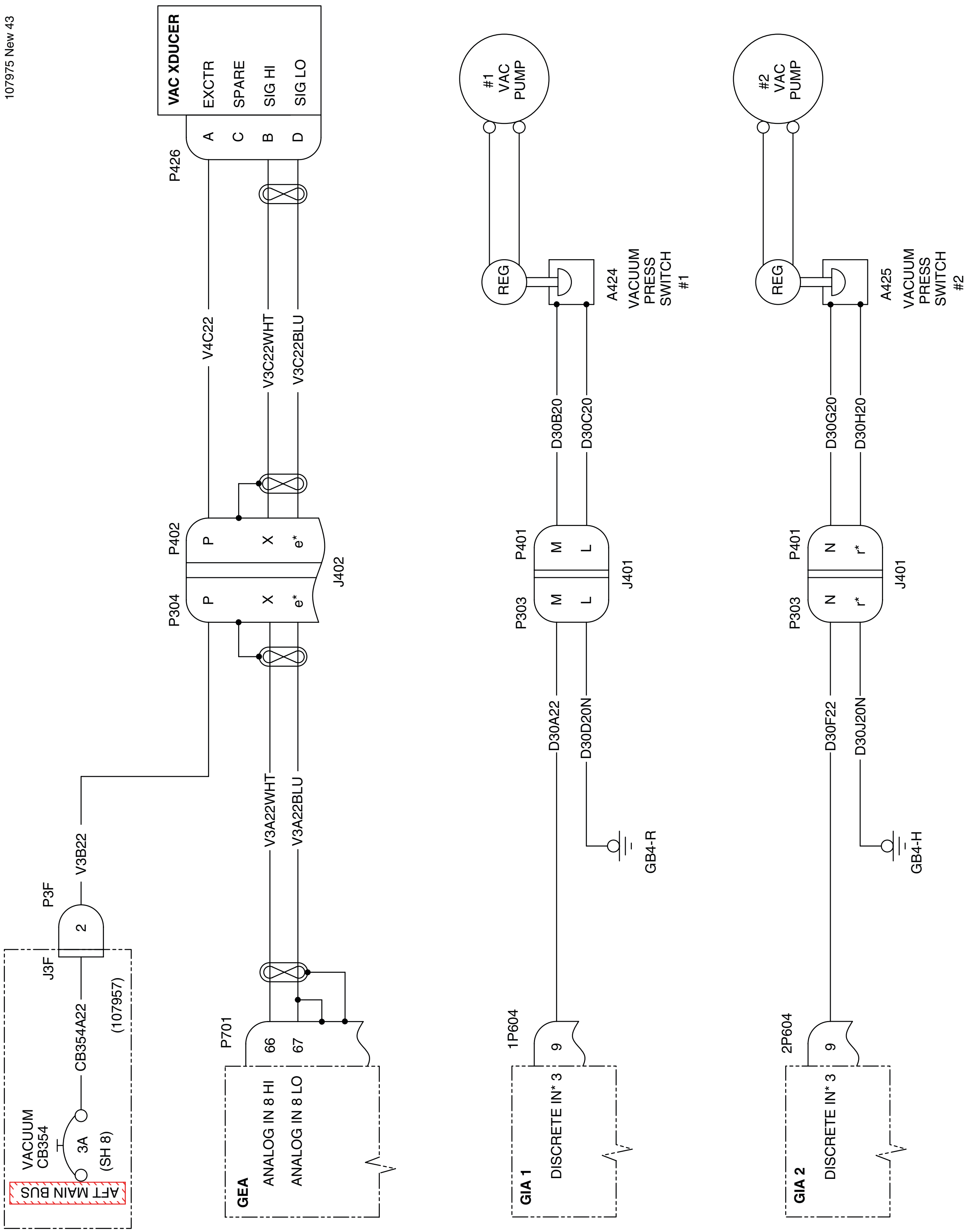
NOTE: VACUUM IS OPTIONAL ON PA46R-350T

Vacuum Indication
Figure 1 (Sheet 2 of 4)



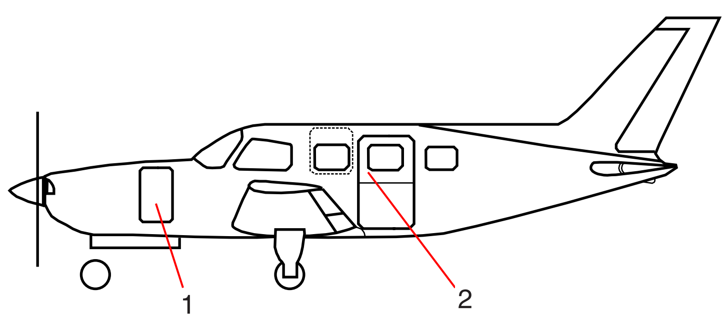
Vacuum Indication
Figure 1 (Sheet 3 of 4)

107975 New 43

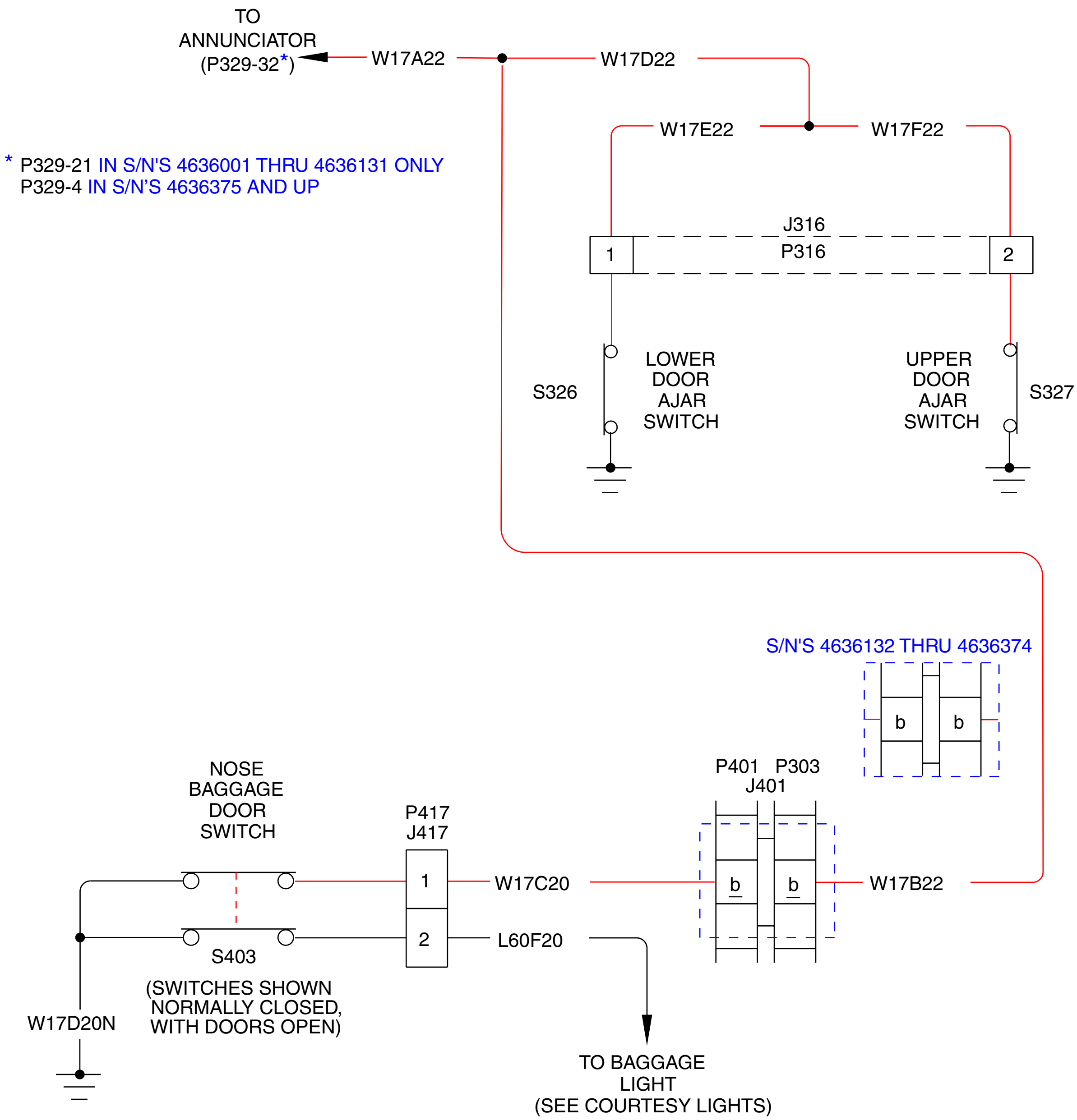


Vacuum Indication
Figure 1 (Sheet 4 of 4)

Item #	Designation	Description
1	S403	Nose Baggage Door Switch
2	S326, S327	Door Ajar Switch (Lower & Upper)



Door Ajar Warning
Figure 1 (Sheet 1 of 2)

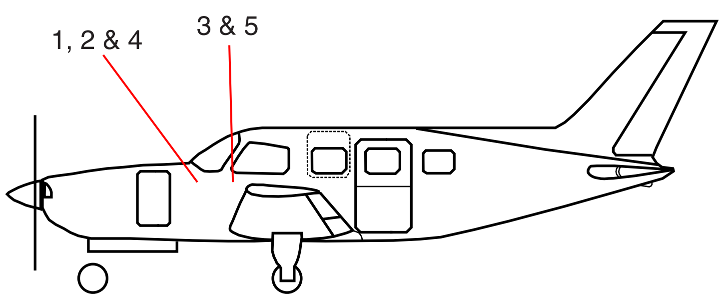


89801 AH 16.0
85508 L
101238 E
101301 C
104051 C
104301 NEW/S
105115 C 15.0

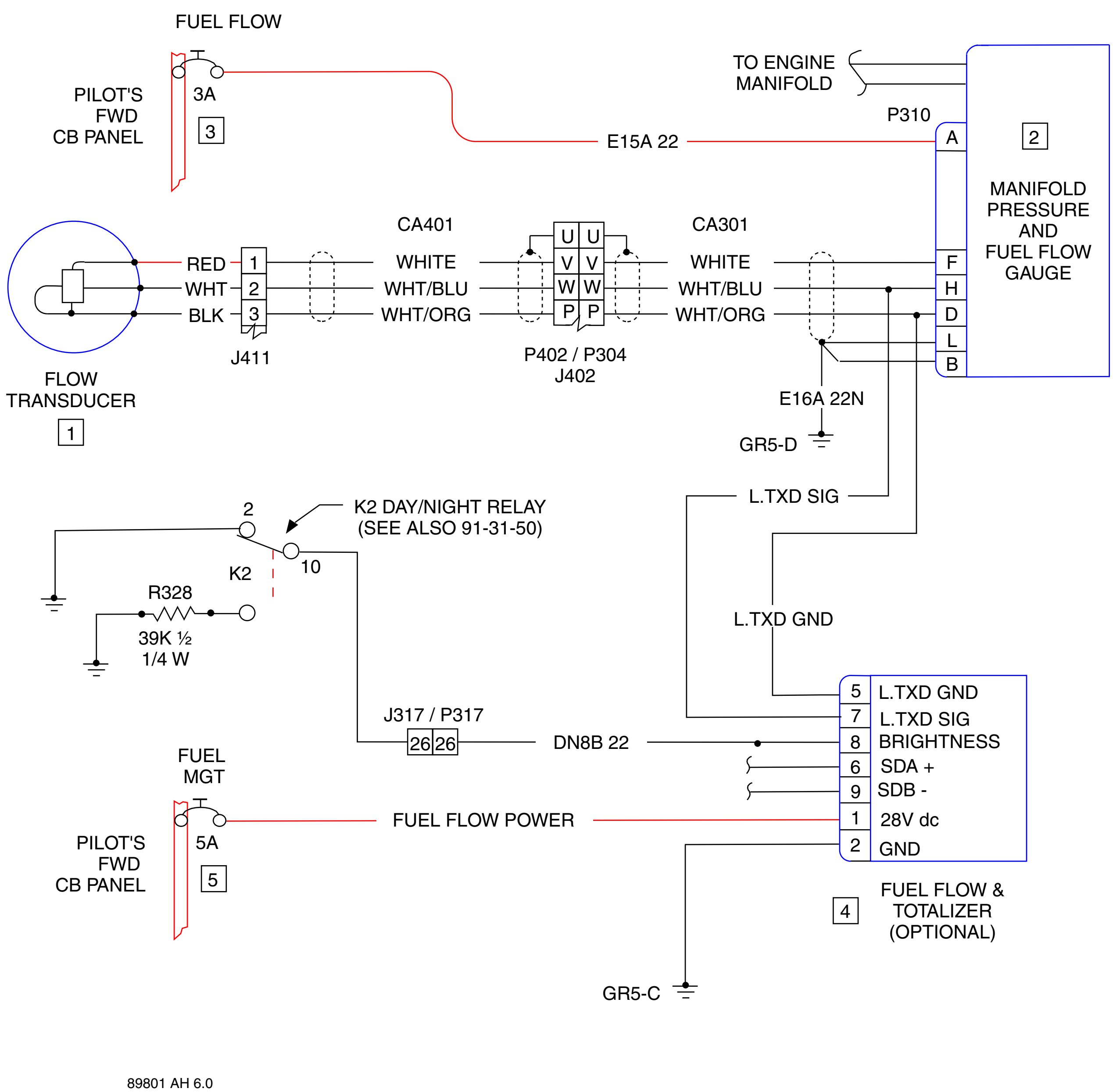
IN S/N'S 4636375 AND UP, SEE ALSO 91-33-20, FIGURE 2, SHEET 3.
IN S/N'S 4692001 AND UP, SEE ALSO 91-33-20, FIGURE 2, SHEET 4.

Door Ajar Warning
Figure 1 (Sheet 2 of 2)

Item #	Designation	Description
1	A413	Flow Transducer
2	M308	Manifold Pressure and Fuel Flow Gauge
3	CB313	Circuit Breaker (3 Amp)
4	M306	Fuel Flow & Totalizer
5	CB324	Circuit Breaker (5 Amp)

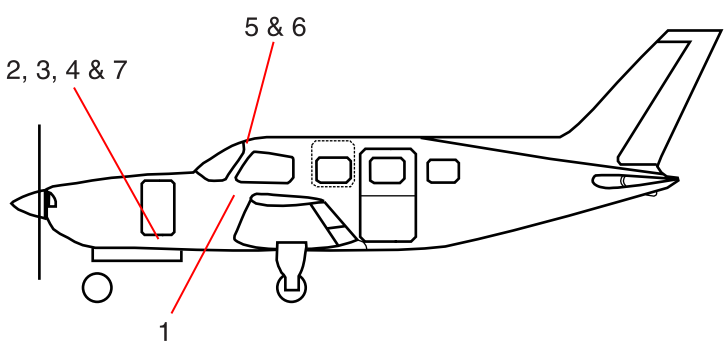


Fuel Flow Gauge(s)
Figure 1 (Sheet 1 of 2)

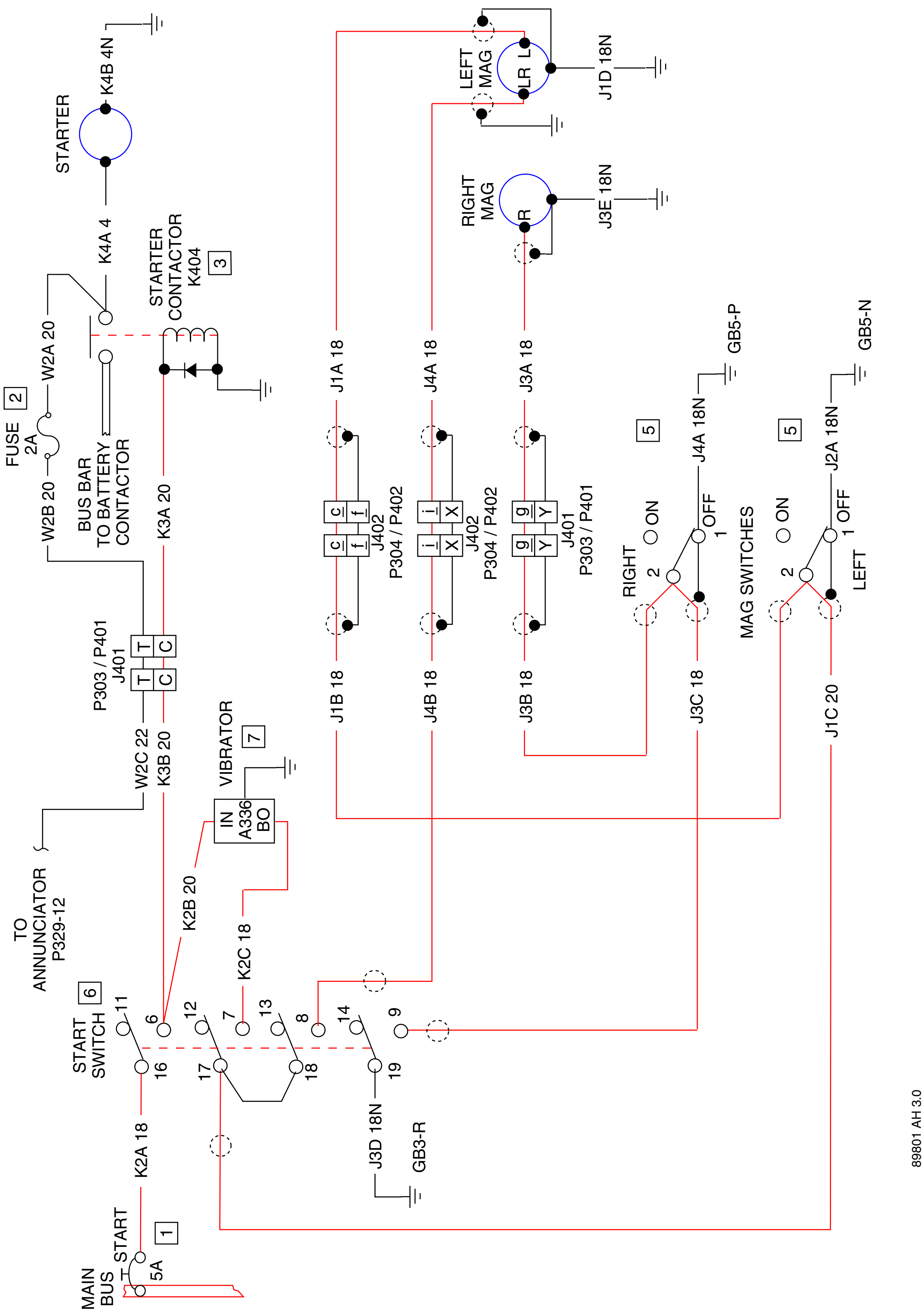


Fuel Flow Gauge(s)
Figure 1 (Sheet 2 of 2)

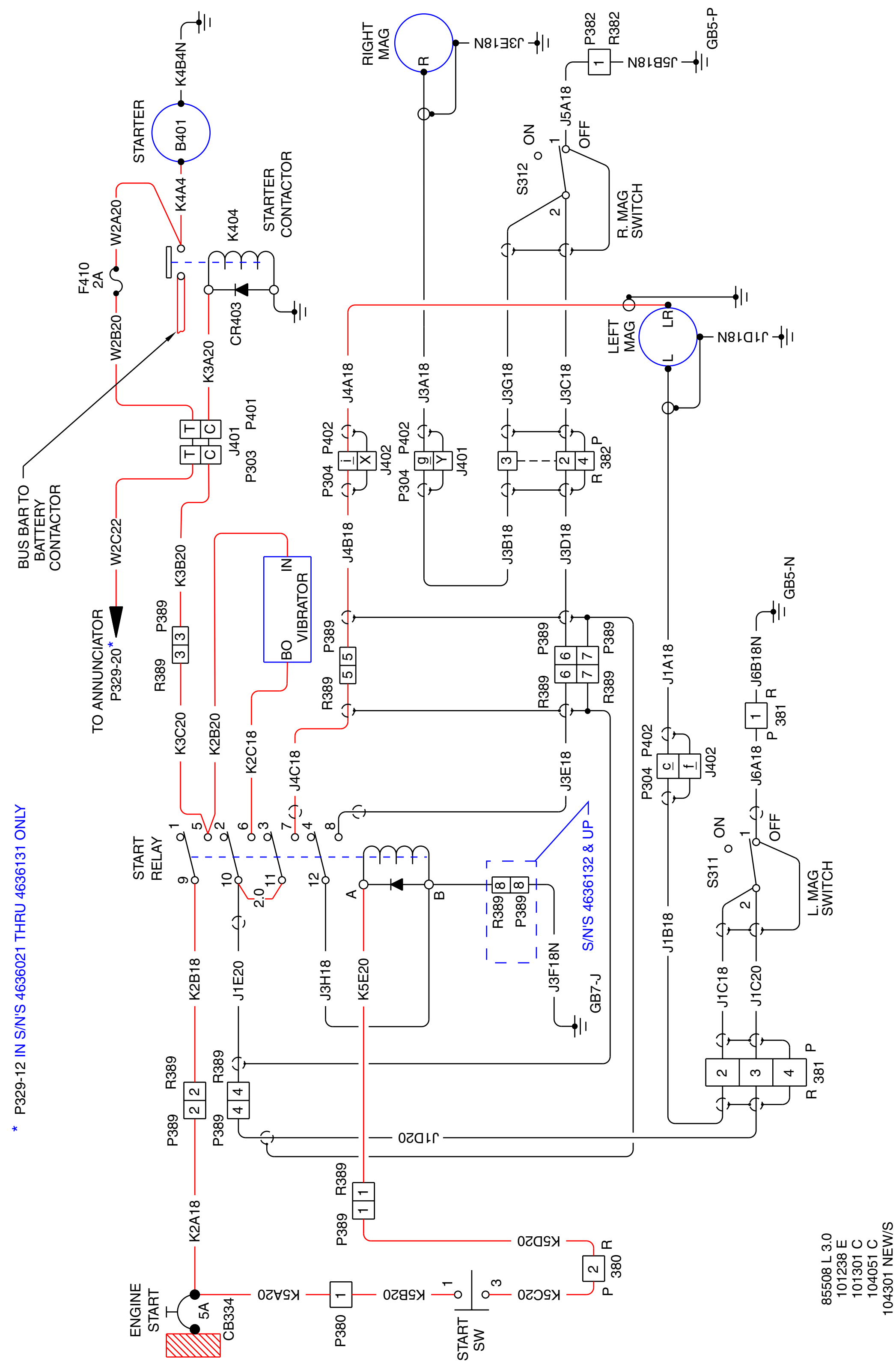
Item #	Designation	Description
1	CB334, CB20	Circuit Breaker - Engine Start (5 Amp)
2	F410	Fuse (2 Amp)
3	K404	Starter Contactor
4	K389	Start Relay
5	S311 (L), S312 (R)	Magneto Switch
6	S380	Start Switch
7	A336	Vibrator



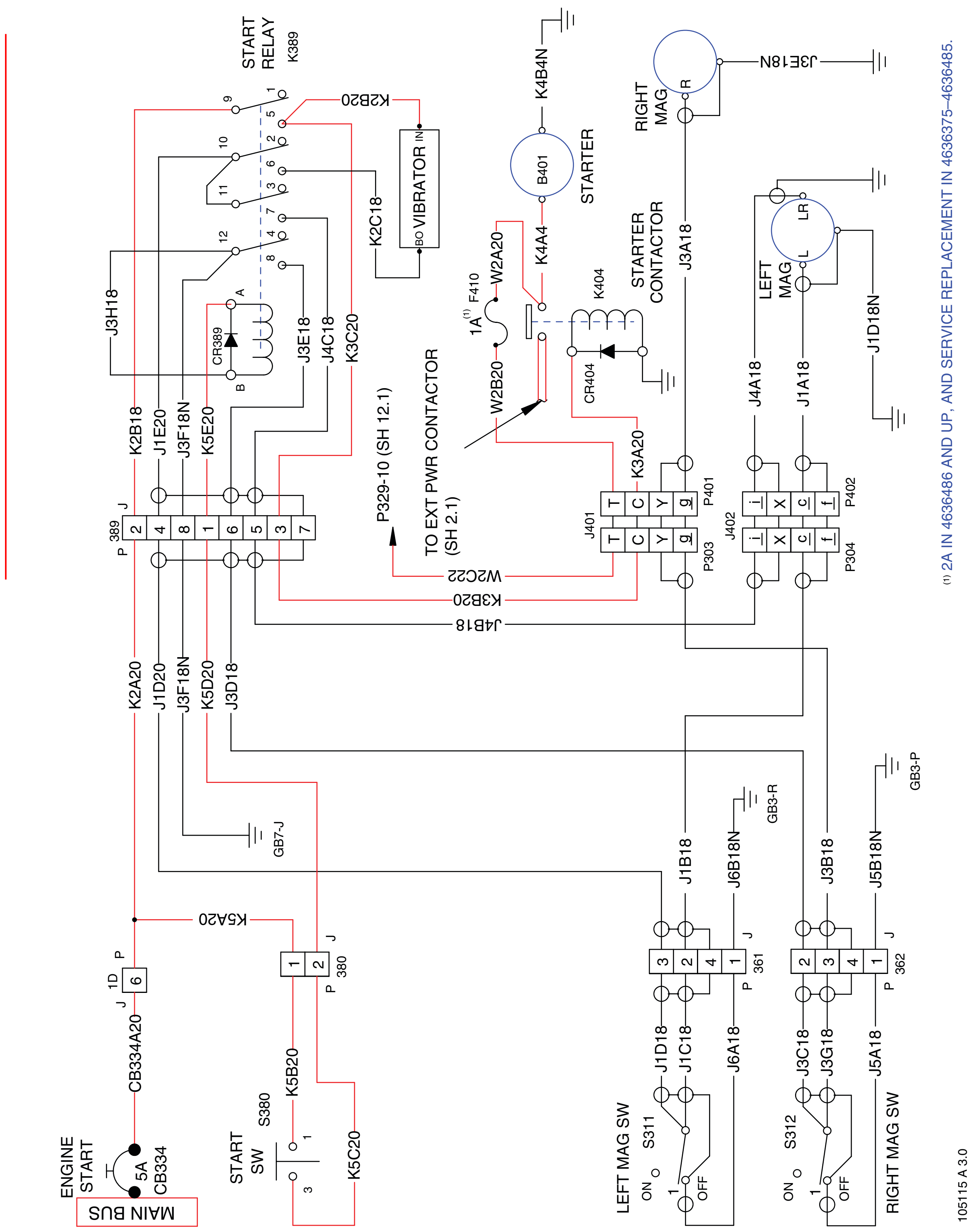
Engine Start and Ignition
Figure 1 (Sheet 1 of 7)



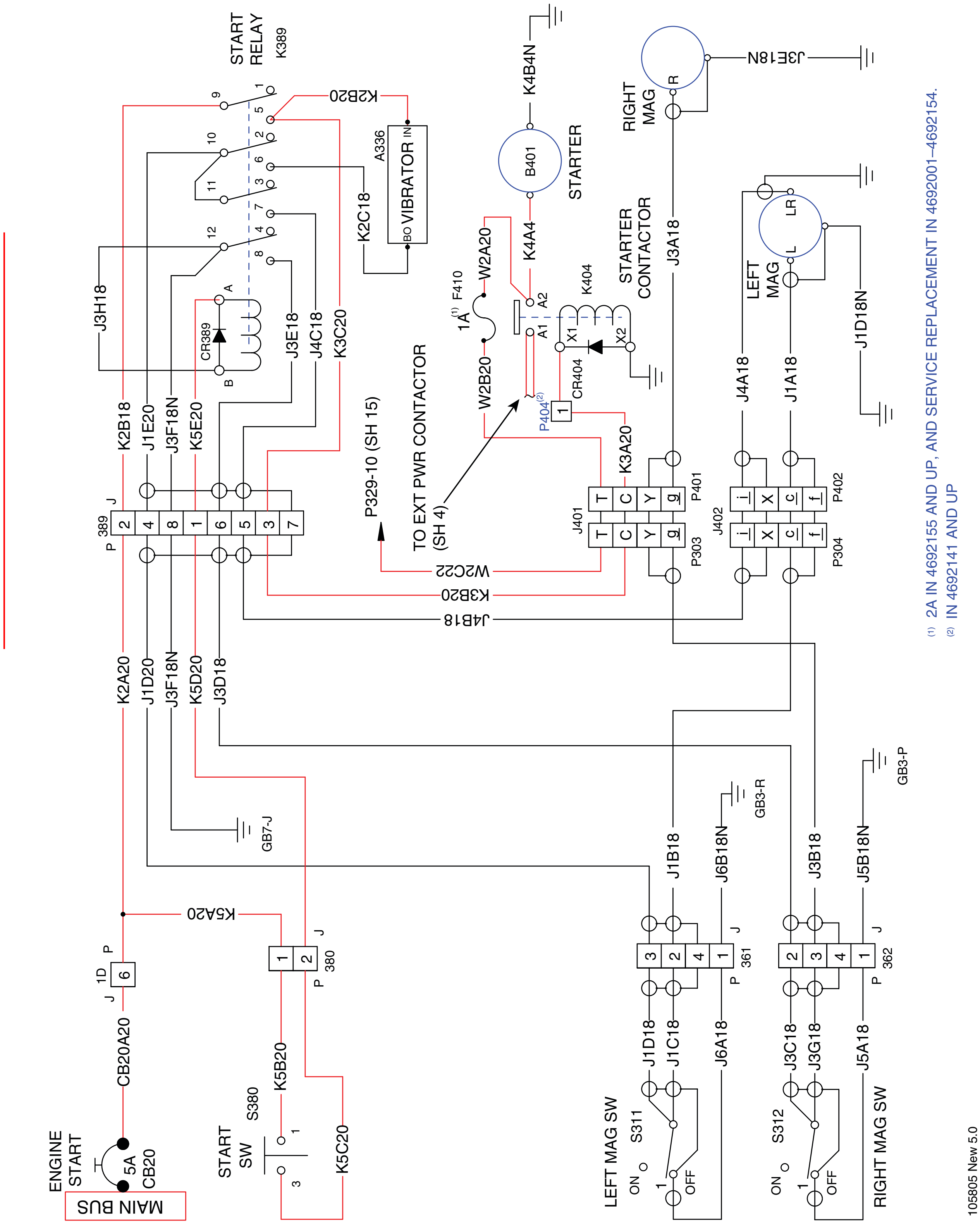
Engine Start and Ignition
Figure 1 (Sheet 2 of 7)



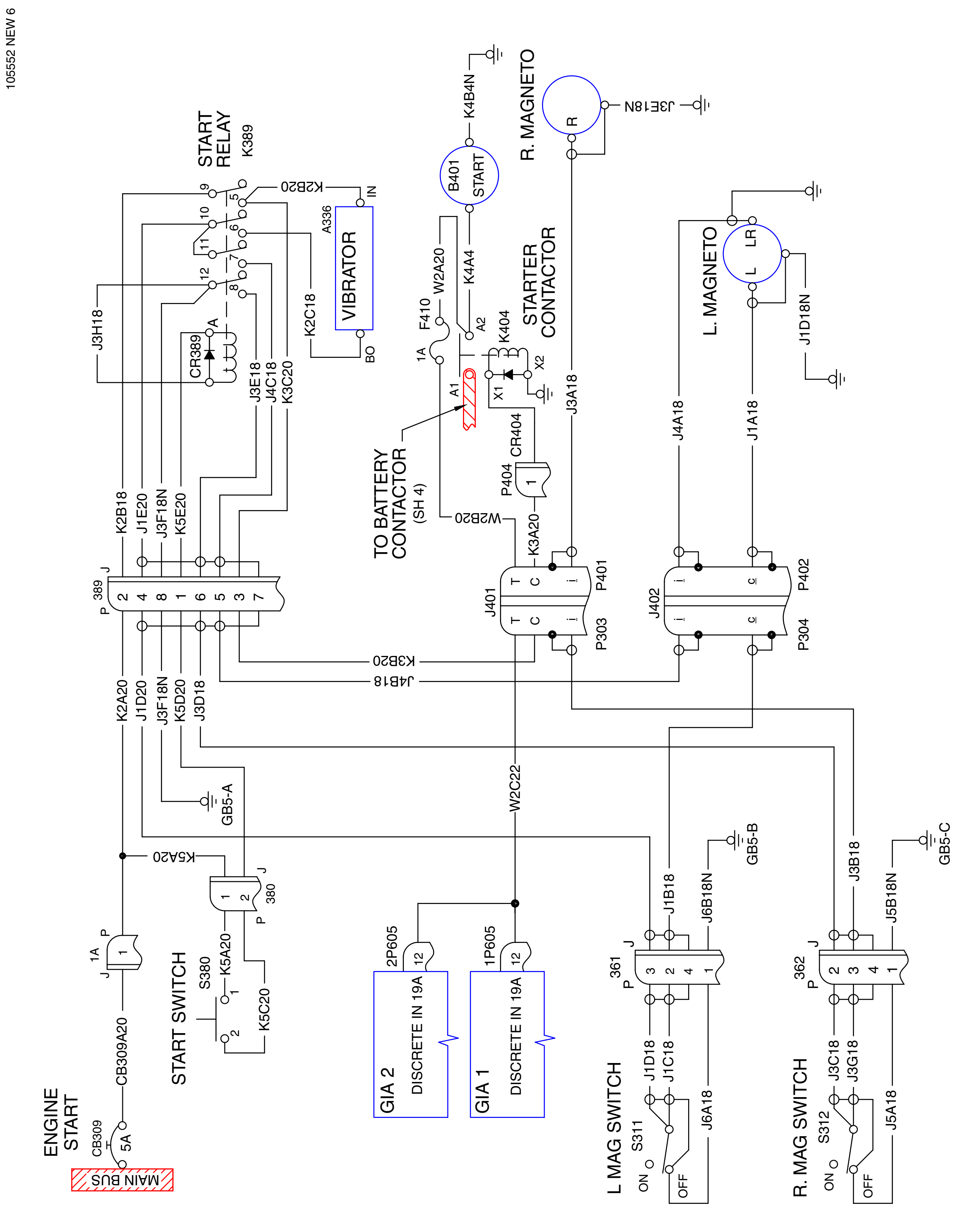
Engine Start and Ignition
Figure 1 (Sheet 3 of 7)



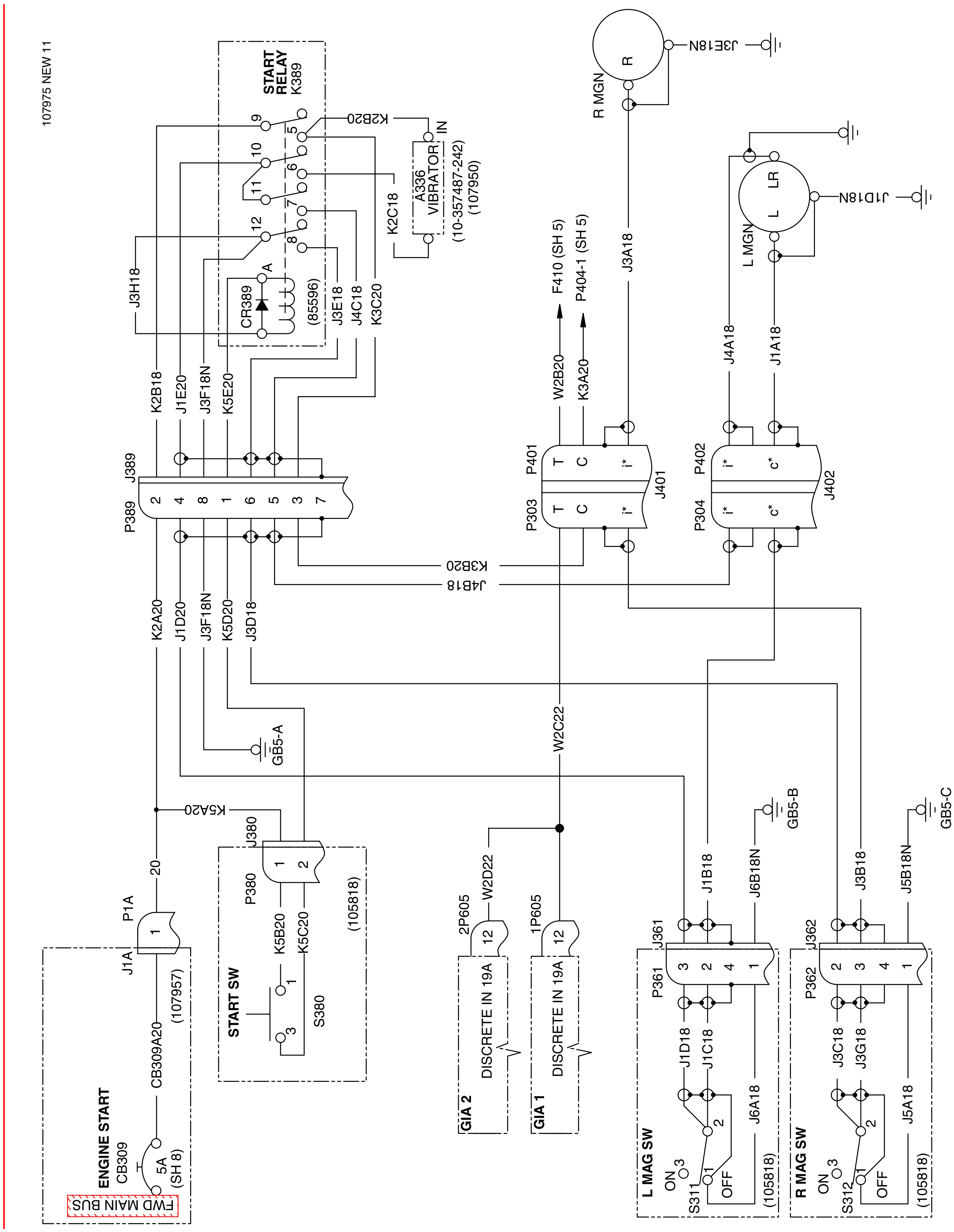
Engine Start and Ignition
Figure 1 (Sheet 4 of 7)



Engine Start and Ignition
Figure 1 (Sheet 5 of 7)



Engine Start and Ignition
Figure 1 (Sheet 6 of 7)



Engine Start and Ignition
Figure 1 (Sheet 7 of 7)

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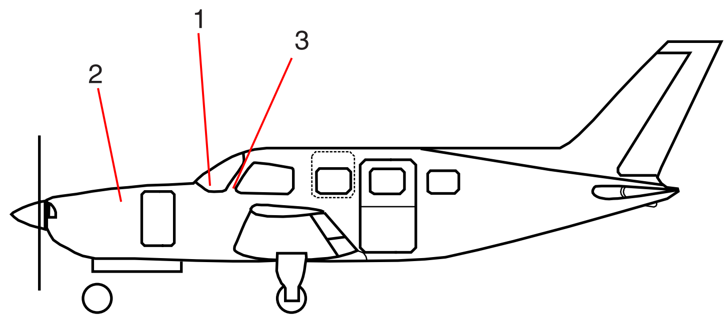
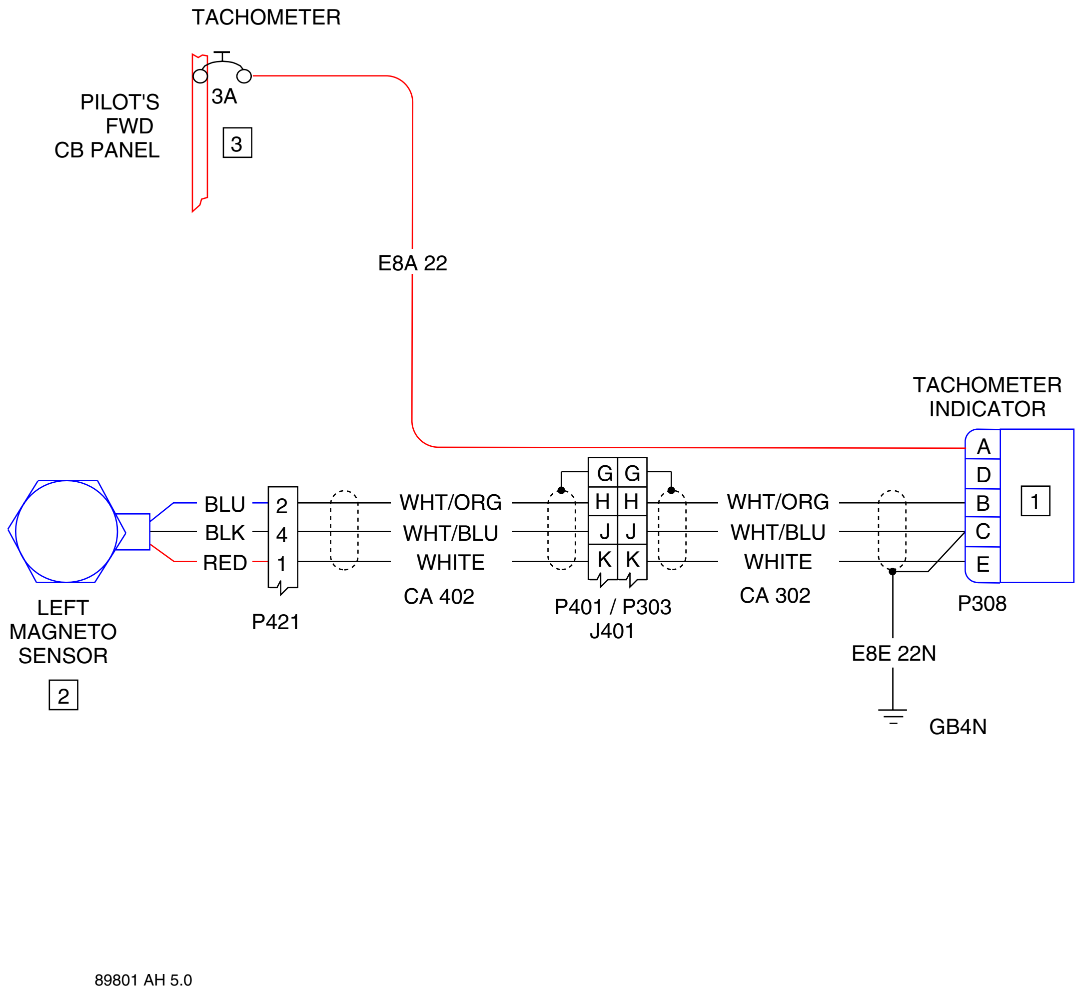
Item #	Designation	Description
1	M304	Tachometer Indicator
2		Magneto Sensor
3	CB314	Circuit Breaker (3 Amp)

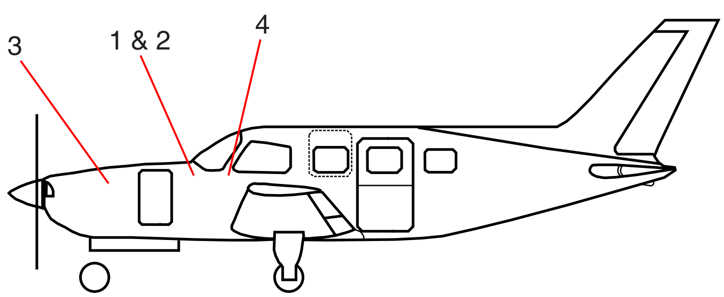


RPM Indicator (Tachometer)
Figure 1 (Sheet 1 of 2)

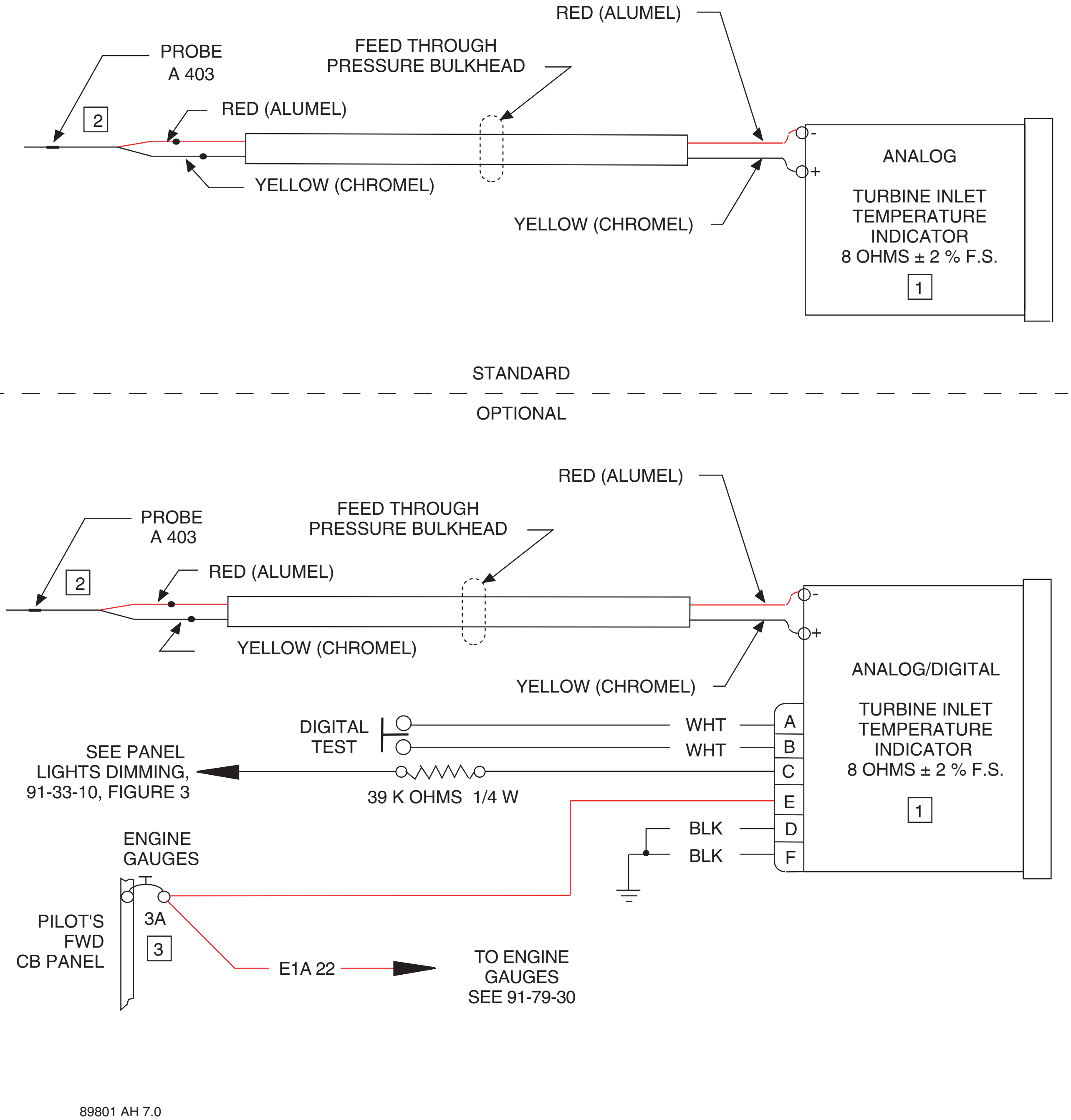


RPM Indicator (Tachometer)
Figure 1 (Sheet 2 of 2)

Item #	Designation	Description
1	M307	Turbine Inlet Temperature Indicator
2	A403	Turbine Inlet Temperature Probe
3	CB315	Circuit Breaker (3 Amp)

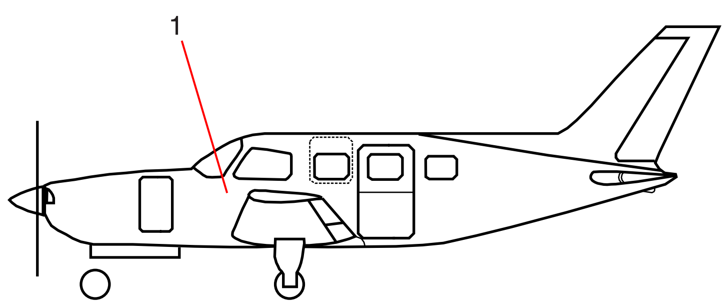


Turbine Inlet Temperature (TIT) Indicator
Figure 1 (Sheet 1 of 2)

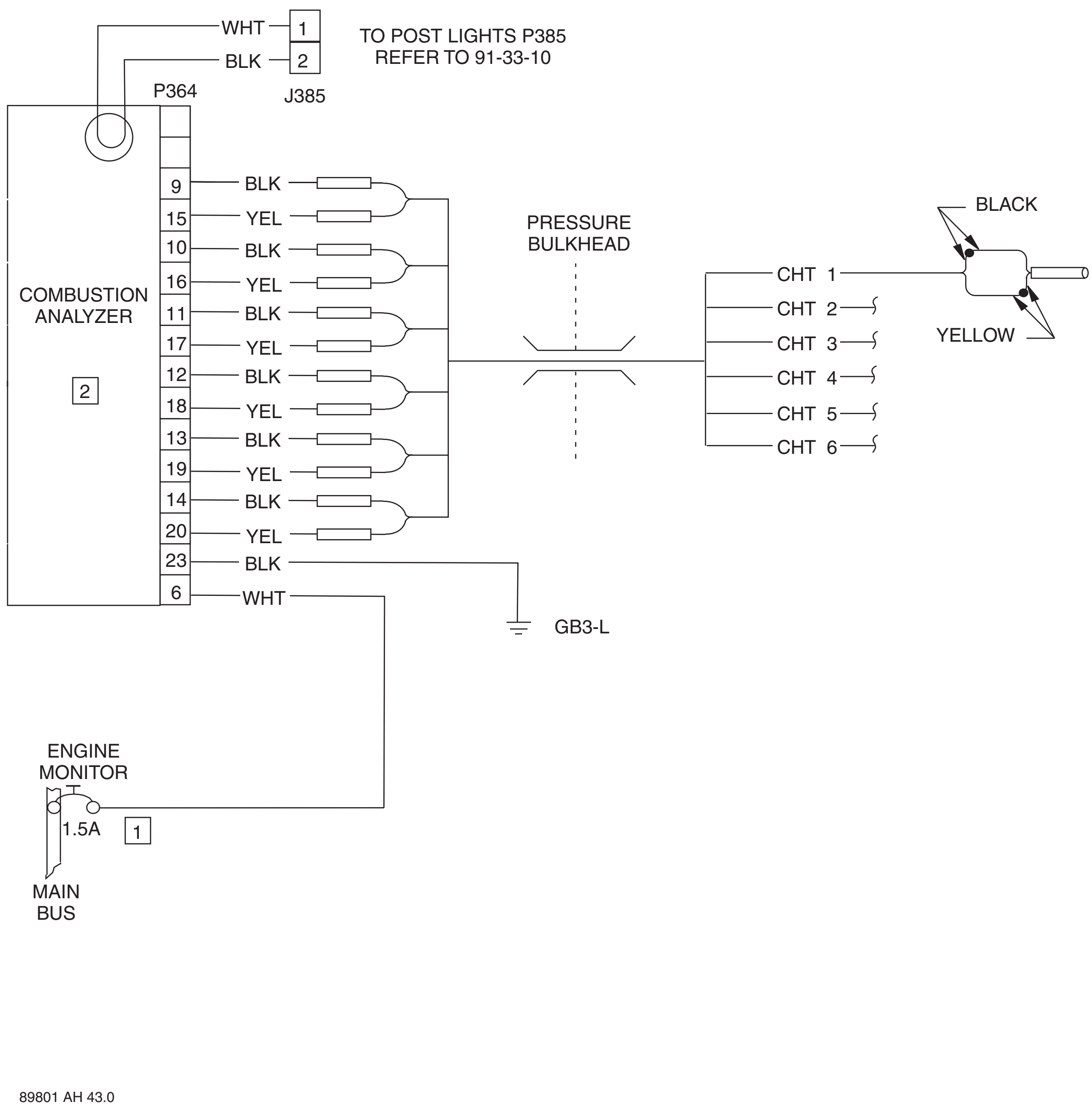


Turbine Inlet Temperature (TIT) Indicator
Figure 1 (Sheet 2 of 2)

Item #	Designation	Description
1	CB316	Circuit Breaker (1.5 Amp)
2	M309	Combustion Analyzer

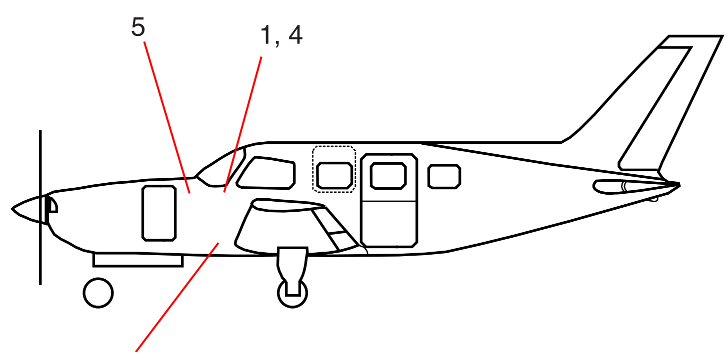


Combustion Analyzer (Optional)
Figure 2 (Sheet 1 of 2)

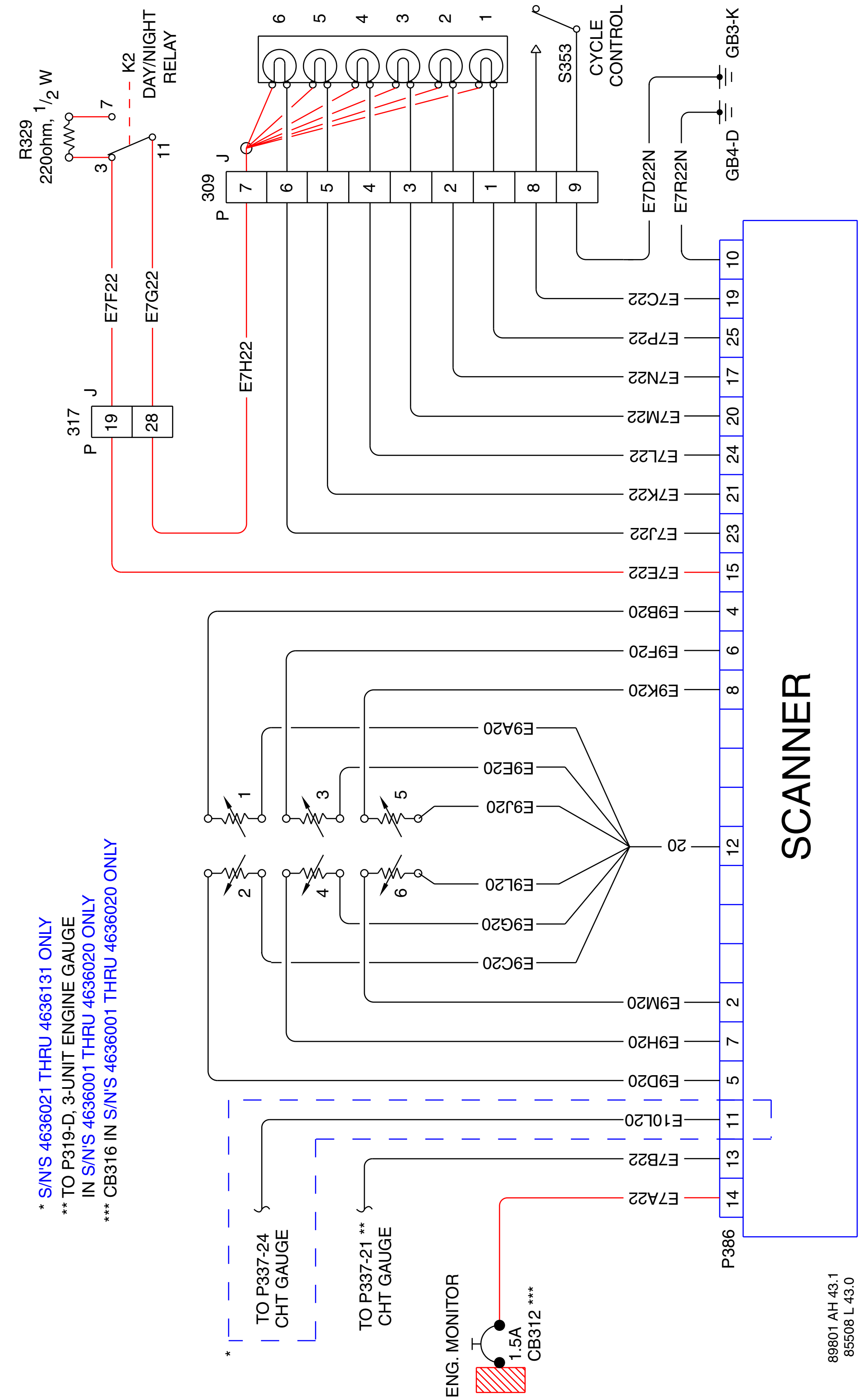


Combustion Analyzer (Optional)
Figure 2 (Sheet 2 of 2)

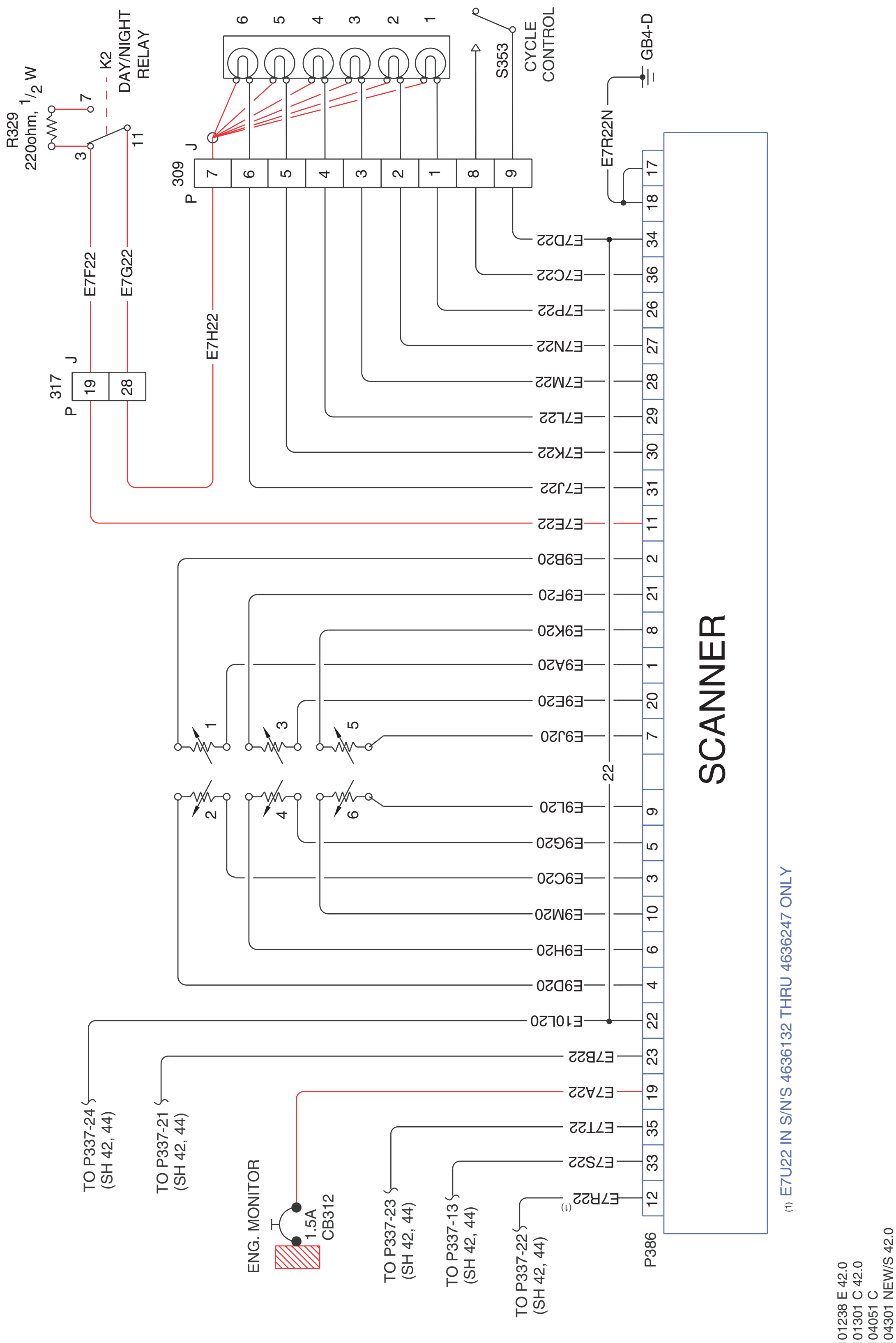
Item #	Designation	Description
1	CB312	Circuit Breaker (1.5 Amp)
2	K2	Day/Night Relay
3	R329	Resistor
4	S353	Cycle Control Switch
5		Scanner



Cylinder Head Temperature (CHT) Indicator
Figure 3 (Sheet 1 of 3)



Cylinder Head Temperature (CHT) Indicator
Figure 3 (Sheet 2 of 3)



Cylinder Head Temperature (CHT) Indicator
Figure 3 (Sheet 3 of 3)

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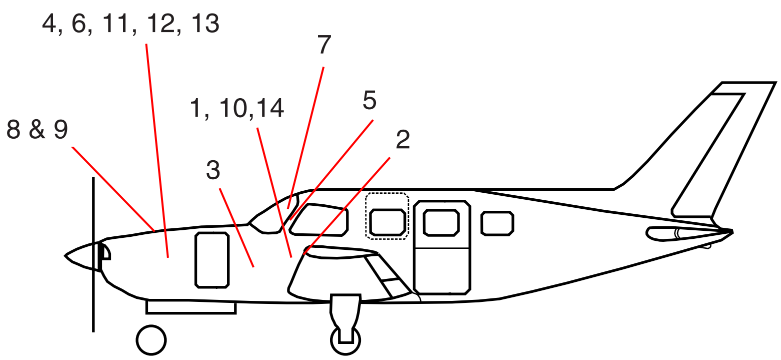
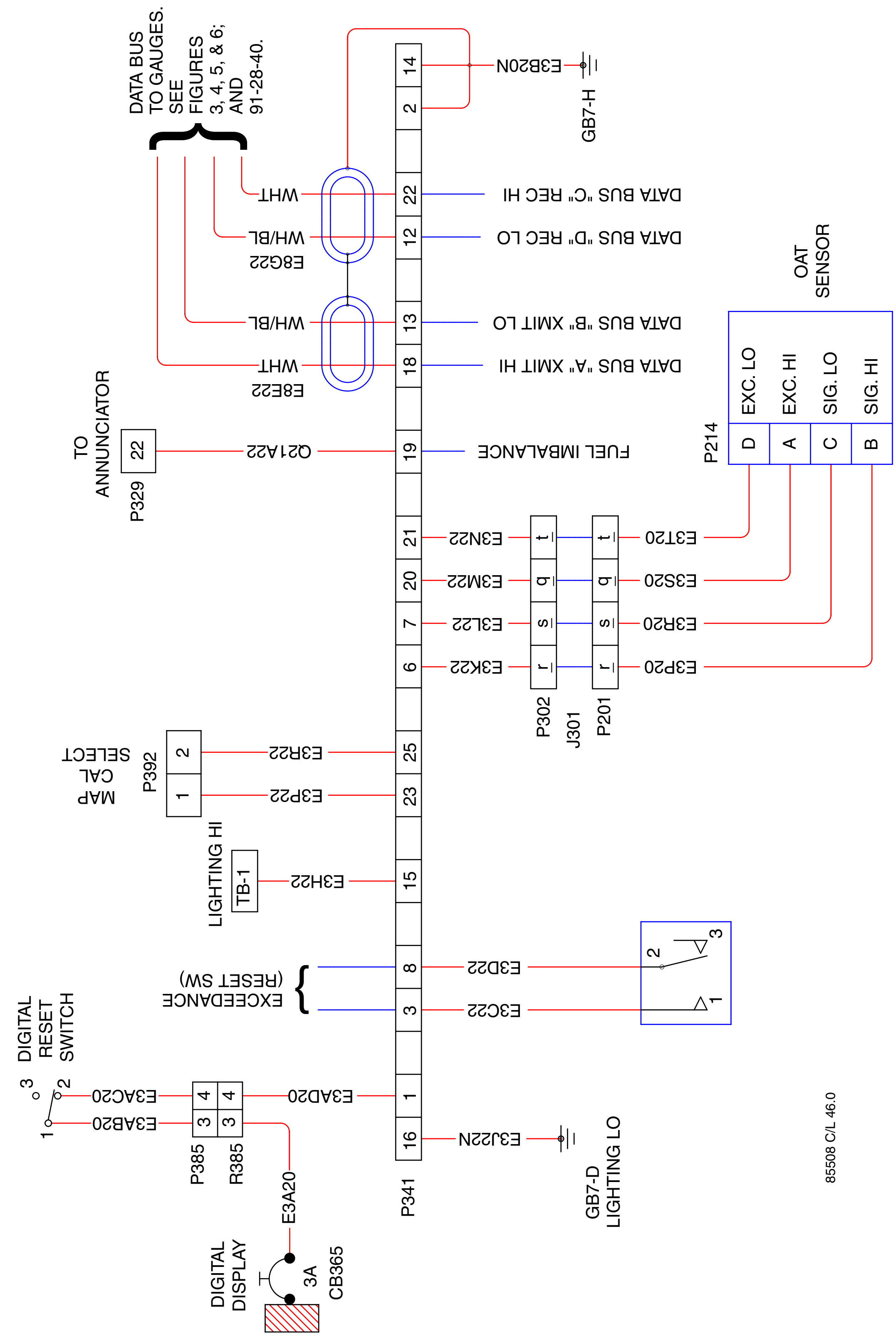
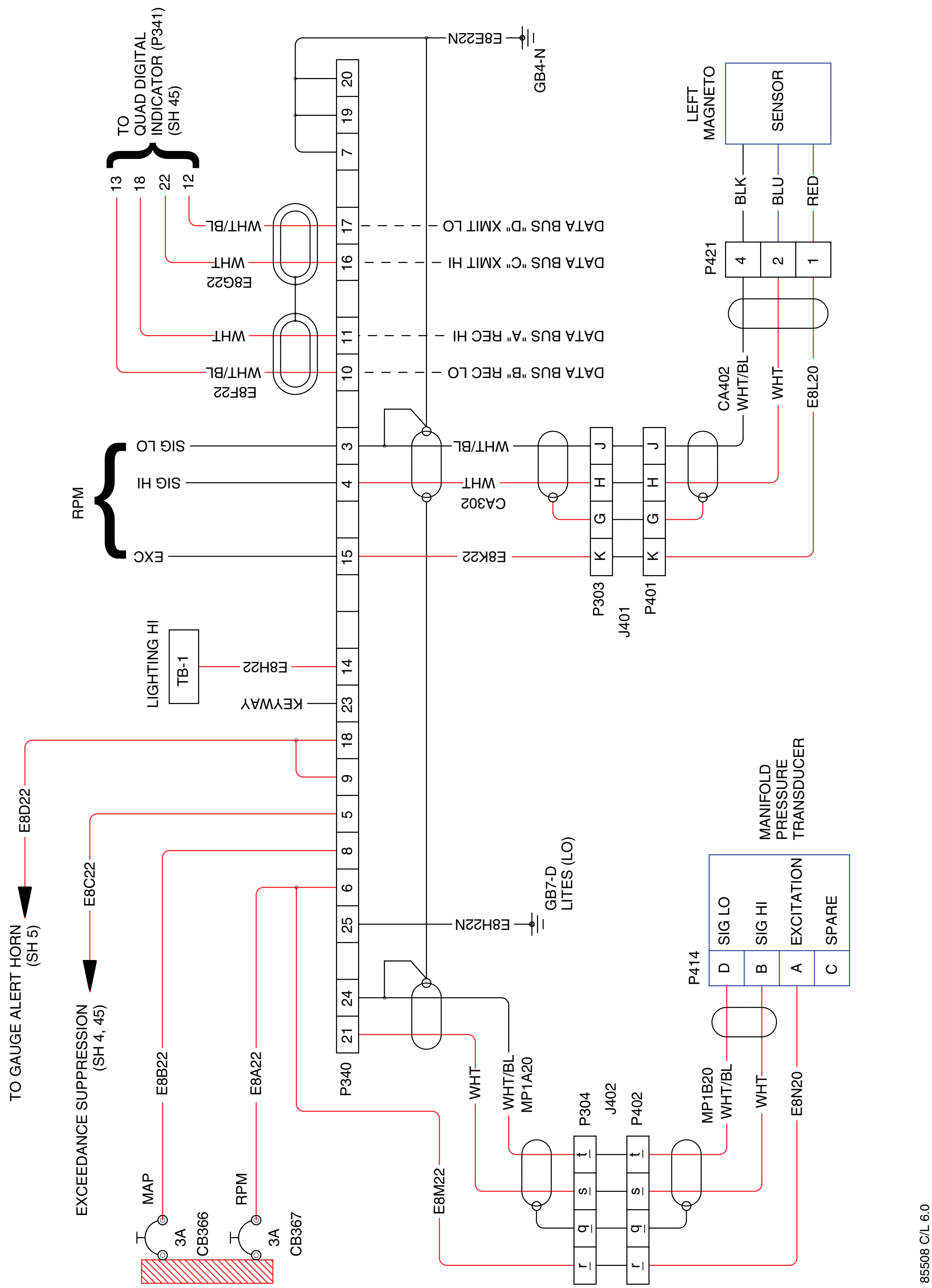
Item #	Designation	Description
1	CB365	Circuit Breaker (3 Amp)
2		OAT Sensor
3		Quad Digital Indicator
4		Magneto Sensor
5	CB366, CB367	Circuit Breaker (3 Amp)
6		Transducer
7	CB368, CB369	Circuit Breaker (3 Amp)
8	A413	Flow Transducer
9		TIT Probe
10	CB313, CB371	Circuit Breaker (3 Amp)
11	A404	Oil Pressure Switch
12	A405	Oil Pressure Transducer
13	A407	Oil Temperature Sensor
14	CB370	Circuit Breaker (3 Amp)



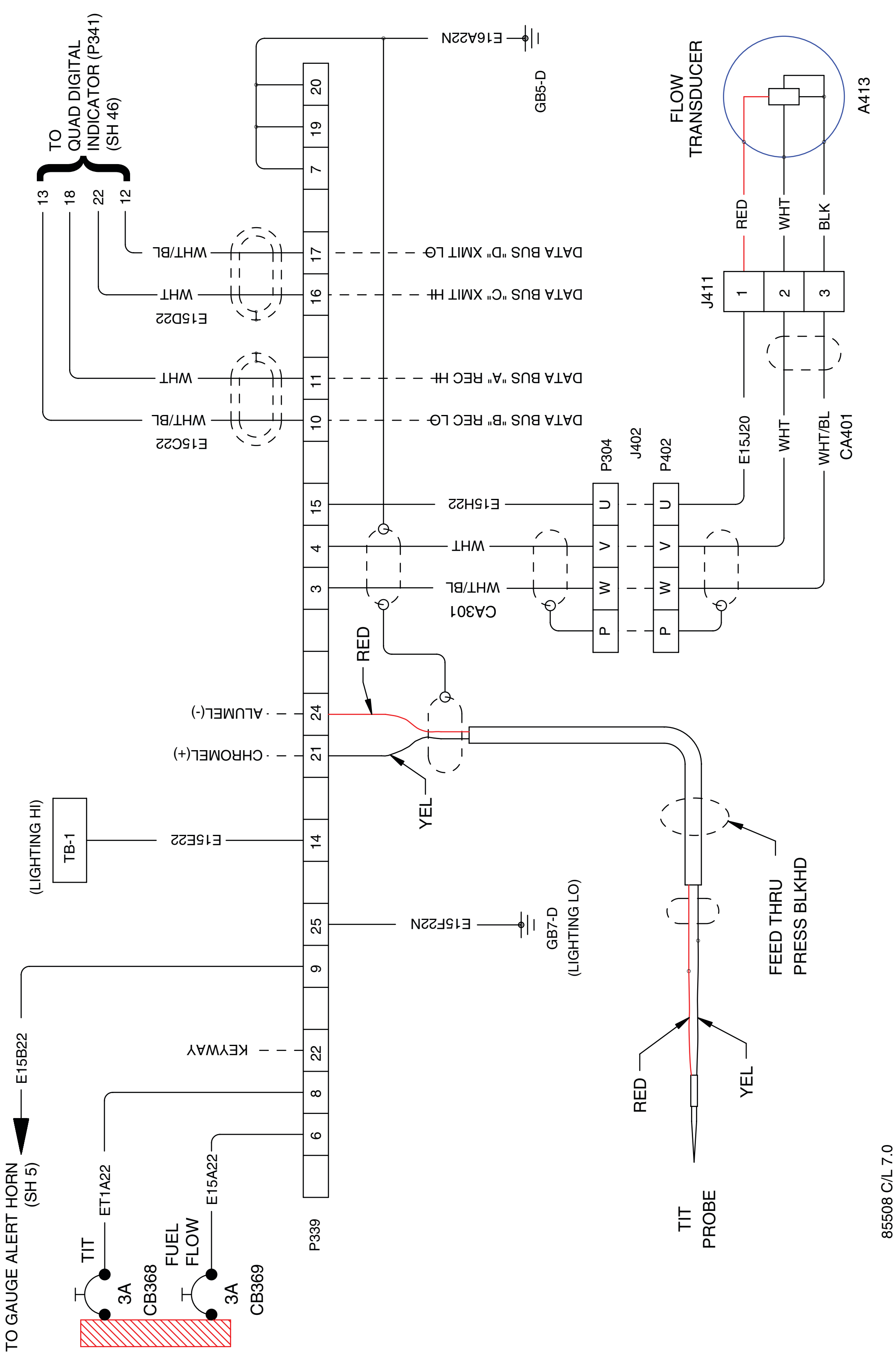
Quad Digital Indicator (QDI)
Figure 1 (Sheet 1 of 6)



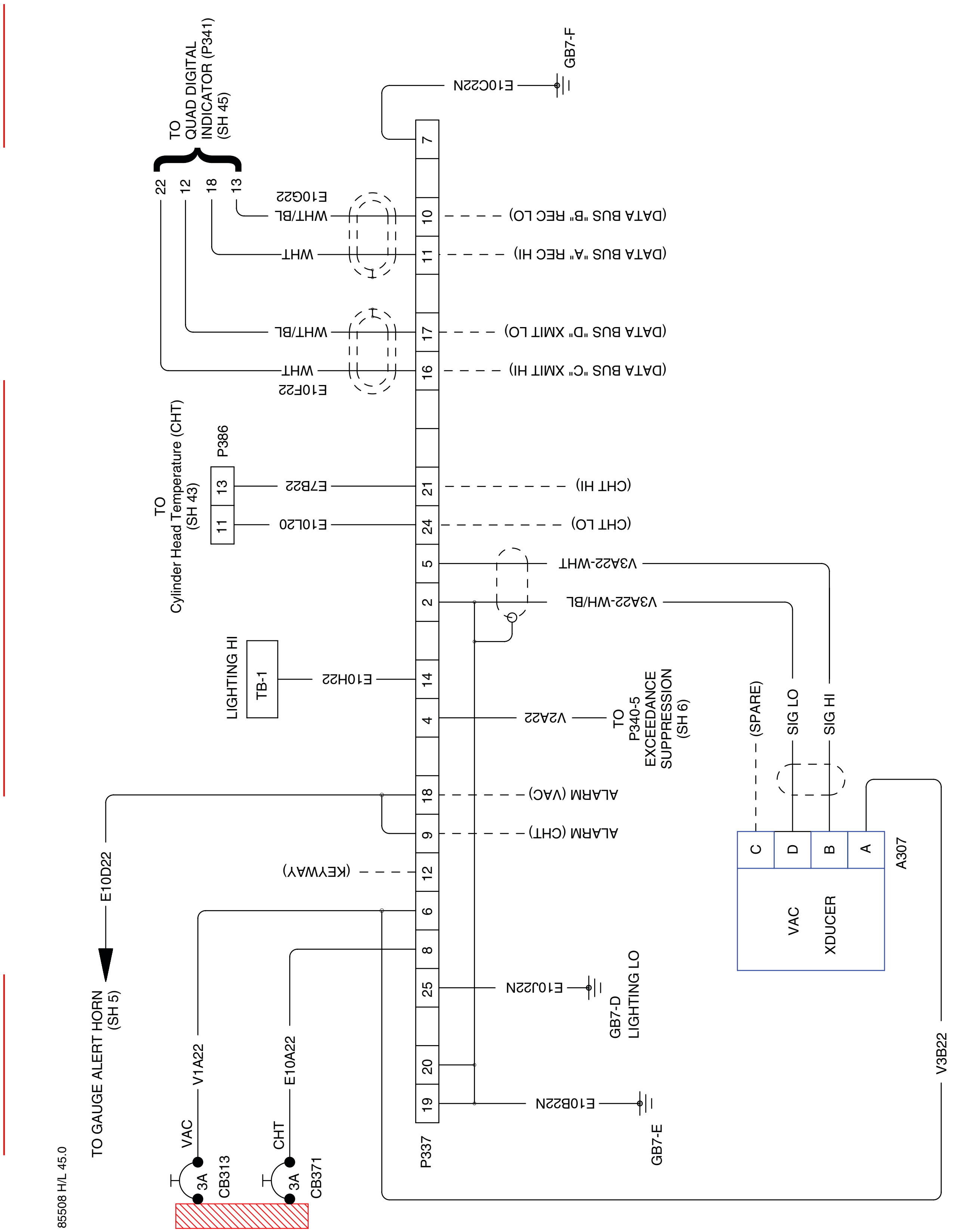
Quad Digital Indicator (QDI)
Figure 1 (Sheet 2 of 6)



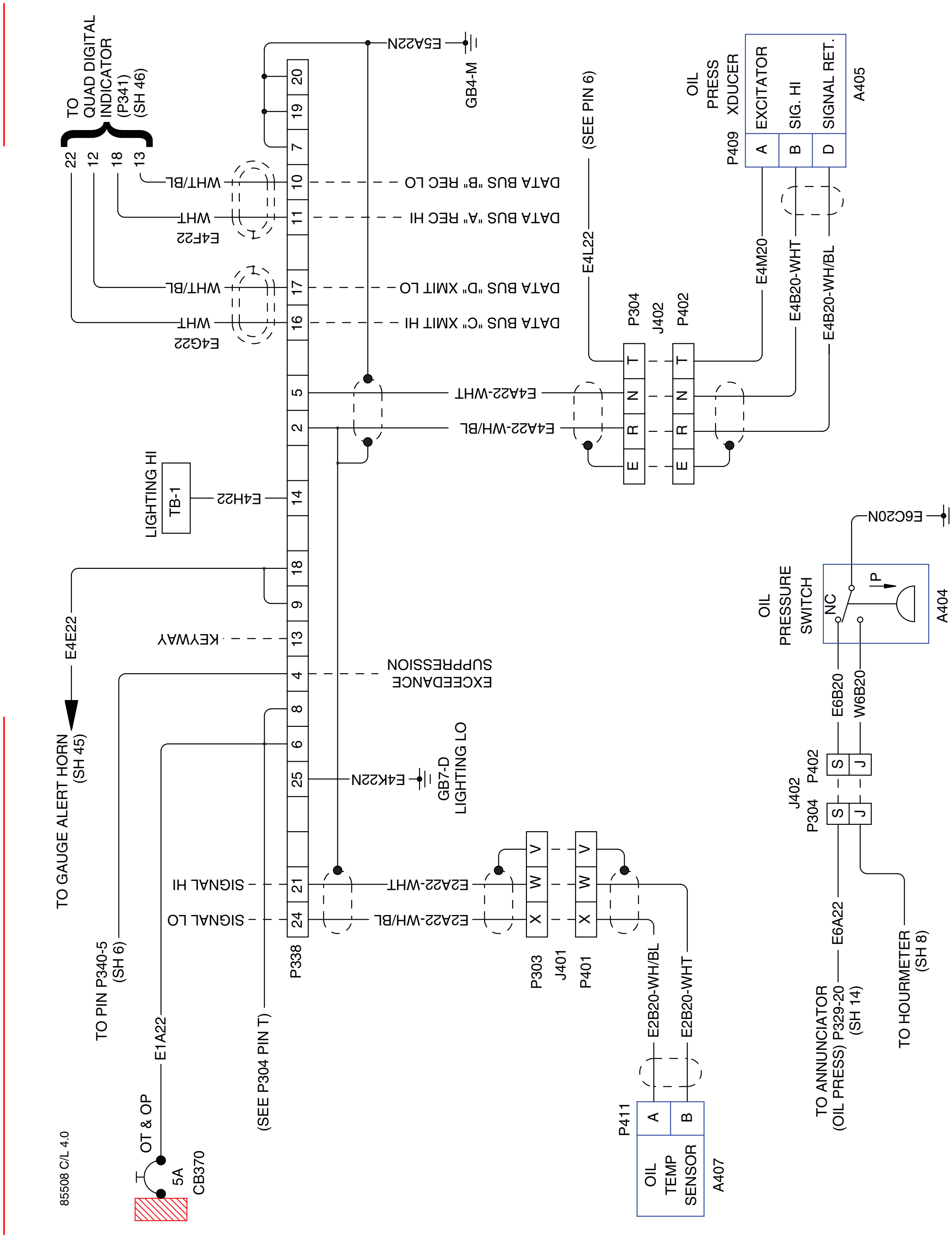
Quad Digital Indicator (QDI)
Figure 1 (Sheet 3 of 6)



Quad Digital Indicator (QDI)
Figure 1 (Sheet 4 of 6)

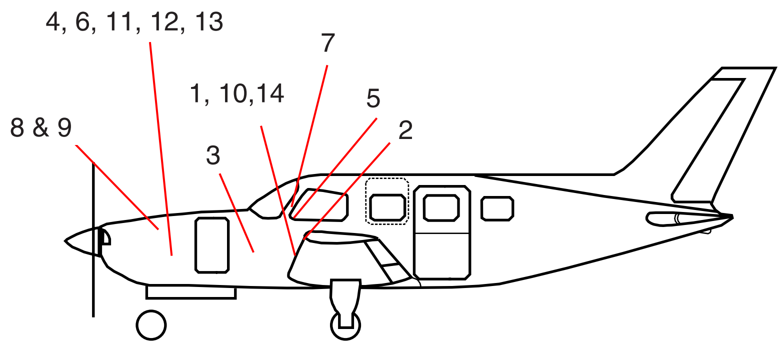


Quad Digital Indicator (QDI)
Figure 1 (Sheet 5 of 6)

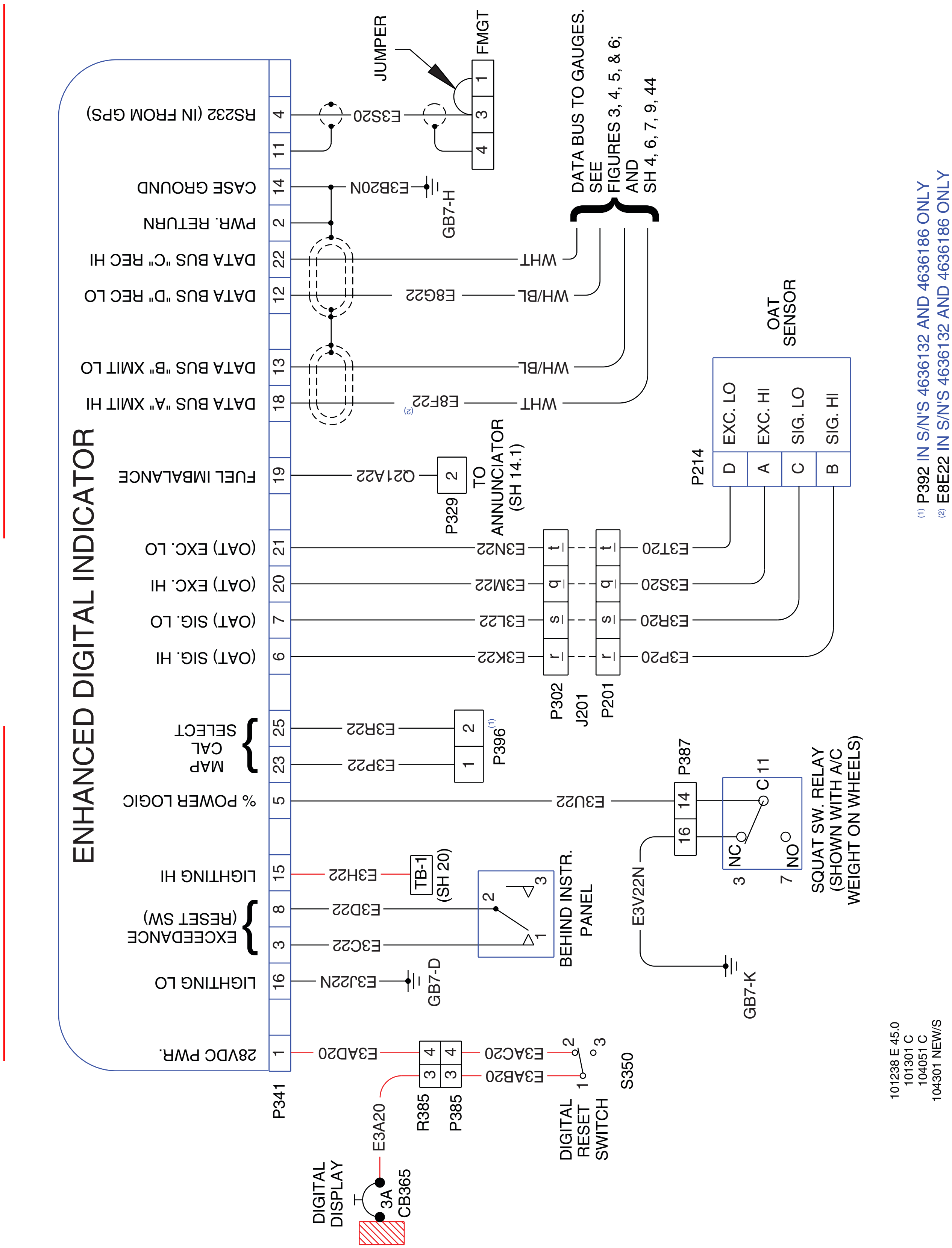


Quad Digital Indicator (QDI)
Figure 1 (Sheet 6 of 6)

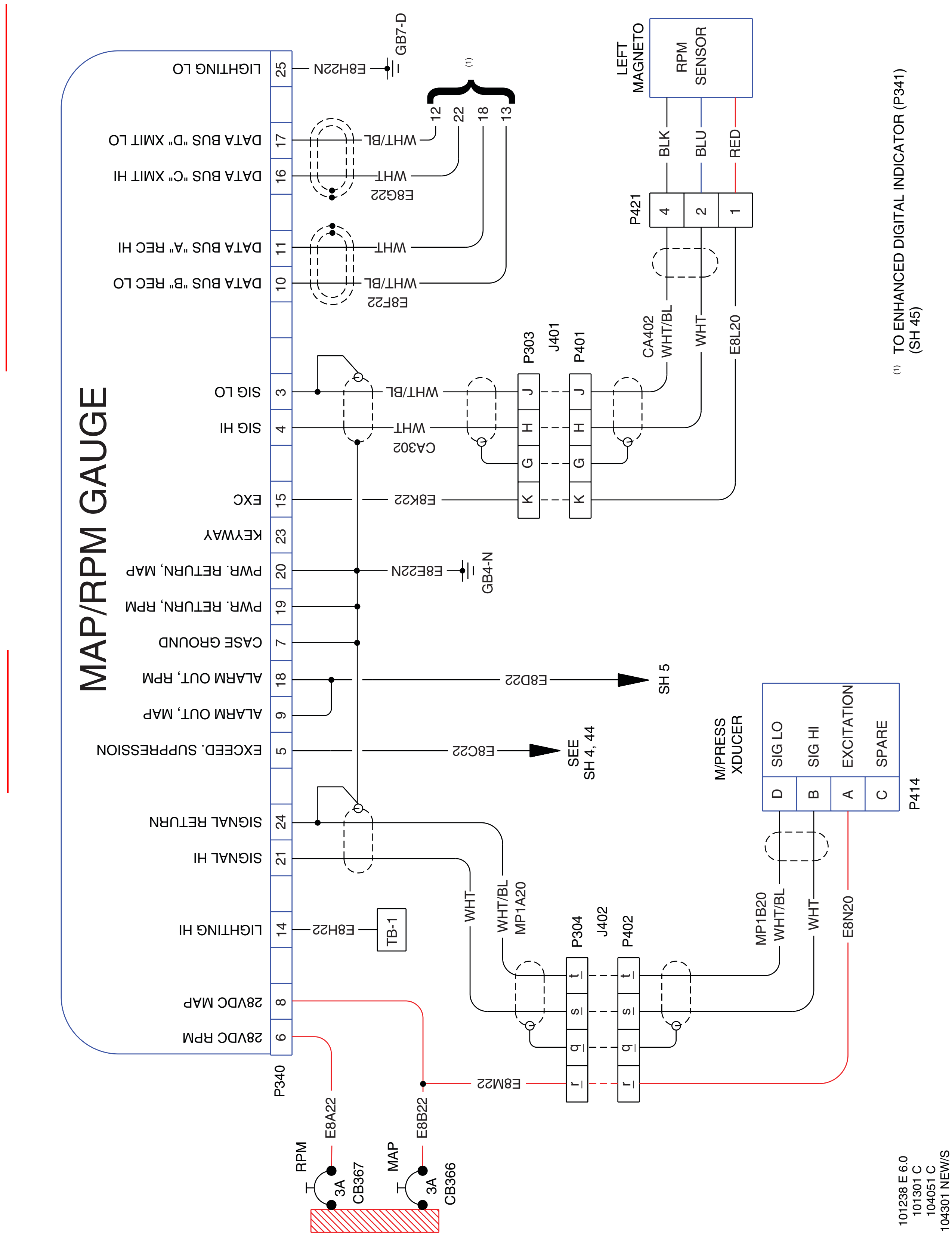
Item #	Designation	Description
1	CB365	Circuit Breaker (3 Amp)
2		OAT Sensor
3		Quad Digital Indicator
4		Magneto Sensor
5	CB366, CB367	Circuit Breaker (3 Amp)
6		Transducer
7	CB368, CB369	Circuit Breaker (3 Amp)
8	A413	Flow Transducer
9		TIT Probe
10	CB313, CB371	Circuit Breaker (3 Amp)
11	A404	Oil Pressure Switch
12	A405	Oil Pressure Transducer
13	A407	Oil Temperature Sensor
14	CB370	Circuit Breaker (3 Amp)



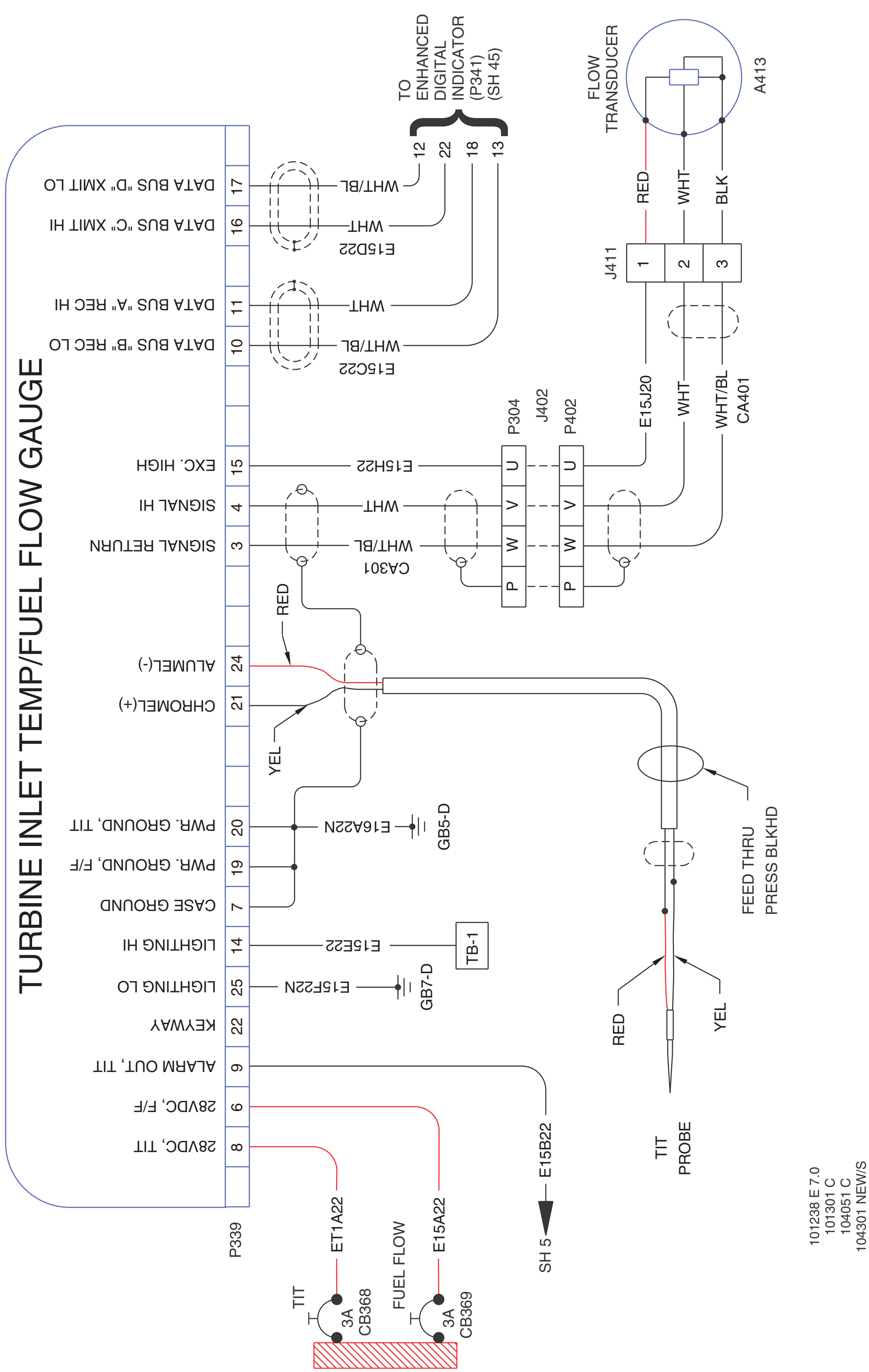
Enhanced Digital Indicator (EDI)
Figure 2 (Sheet 1 of 6)



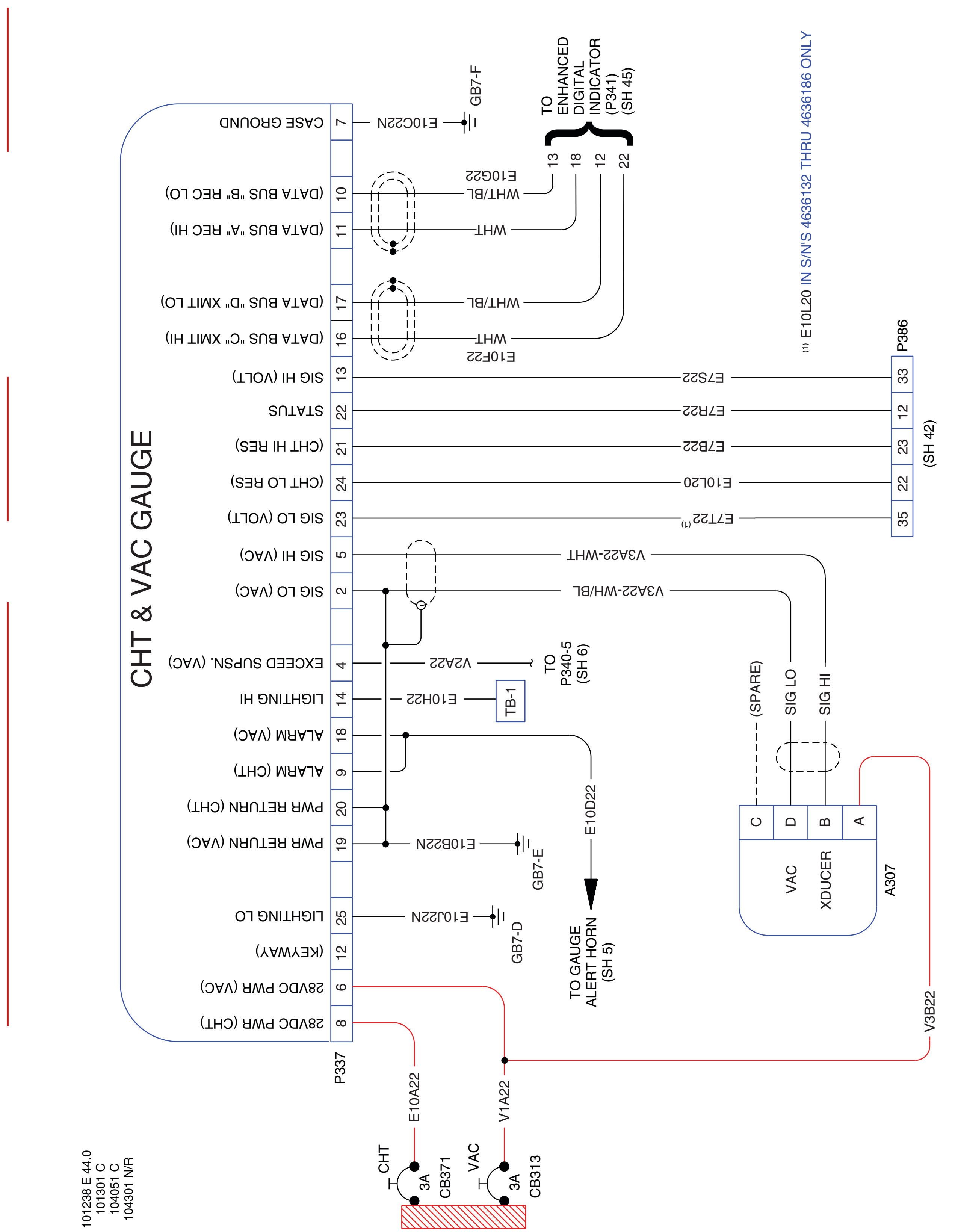
Enhanced Digital Indicator (EDI)
Figure 2 (Sheet 2 of 6)



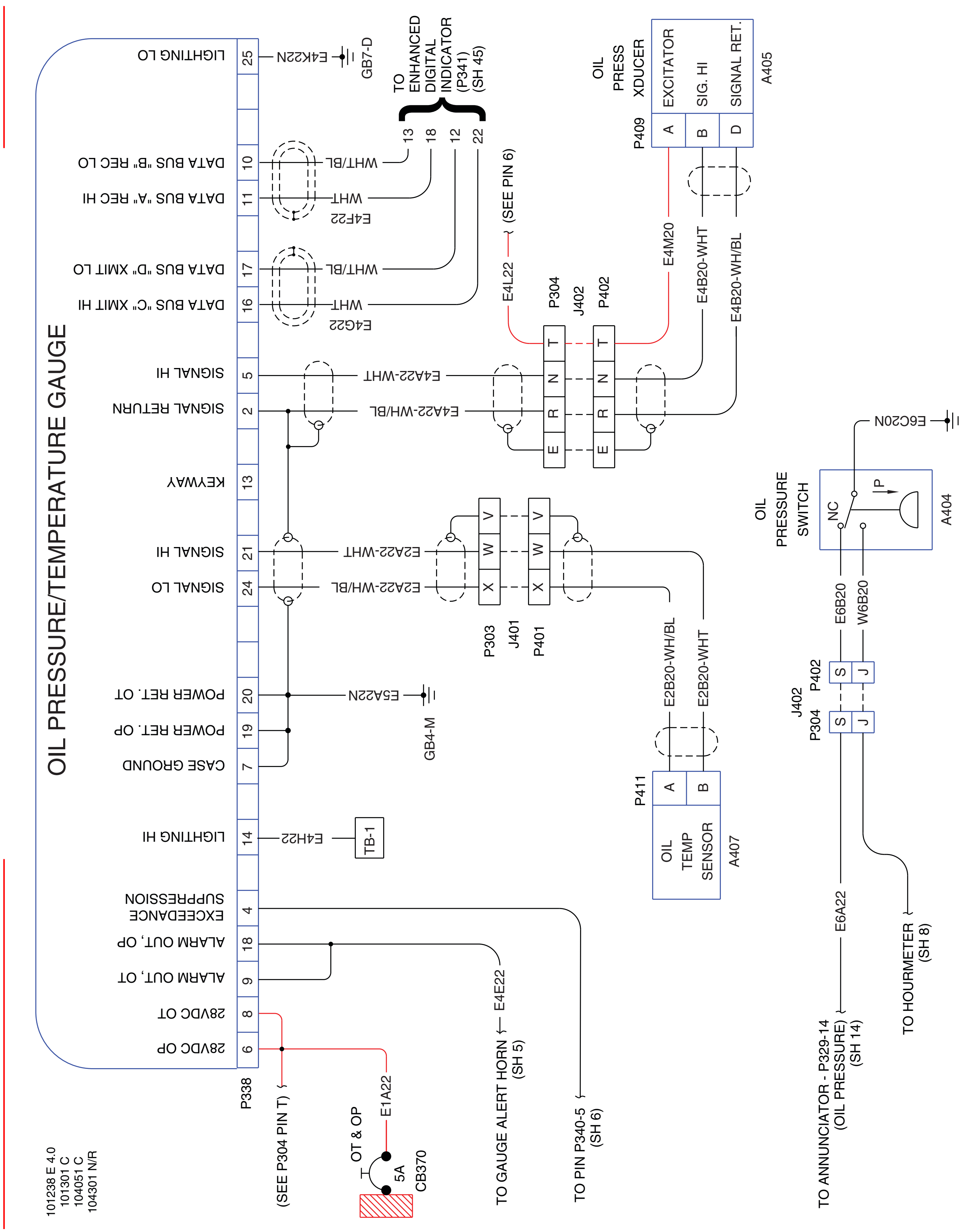
Enhanced Digital Indicator (EDI)
Figure 2 (Sheet 3 of 6)



Enhanced Digital Indicator (EDI)
Figure 2 (Sheet 4 of 6)

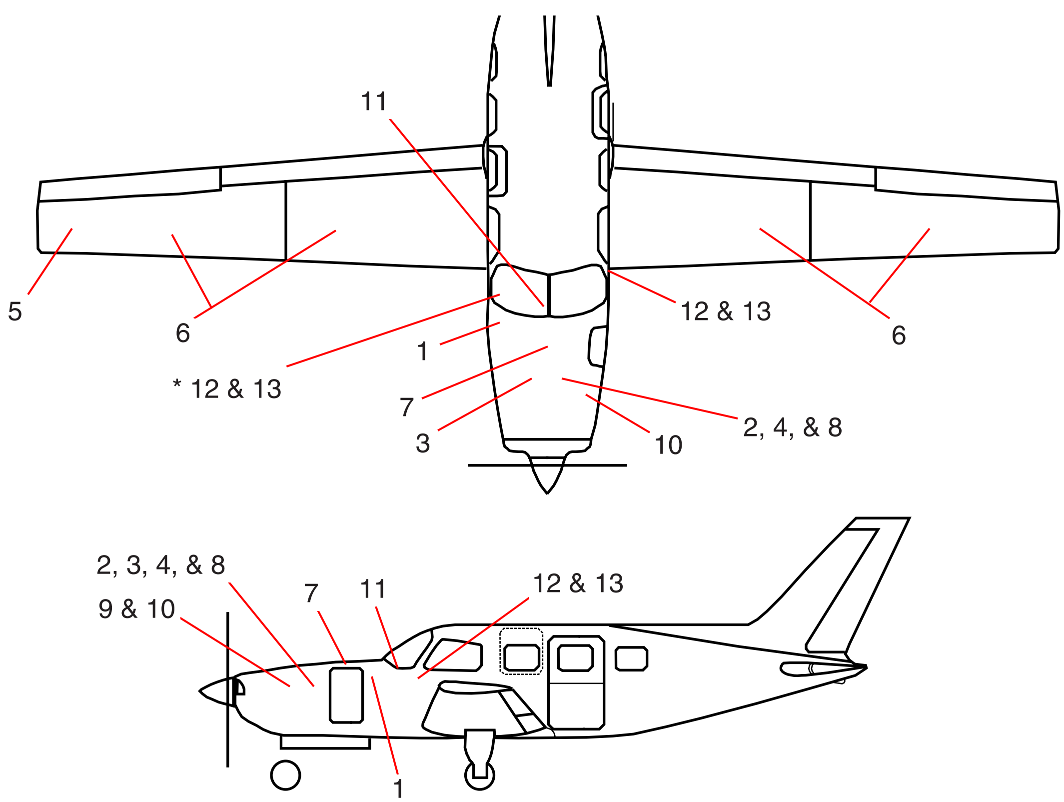


Enhanced Digital Indicator (EDI)
Figure 2 (Sheet 5 of 6)

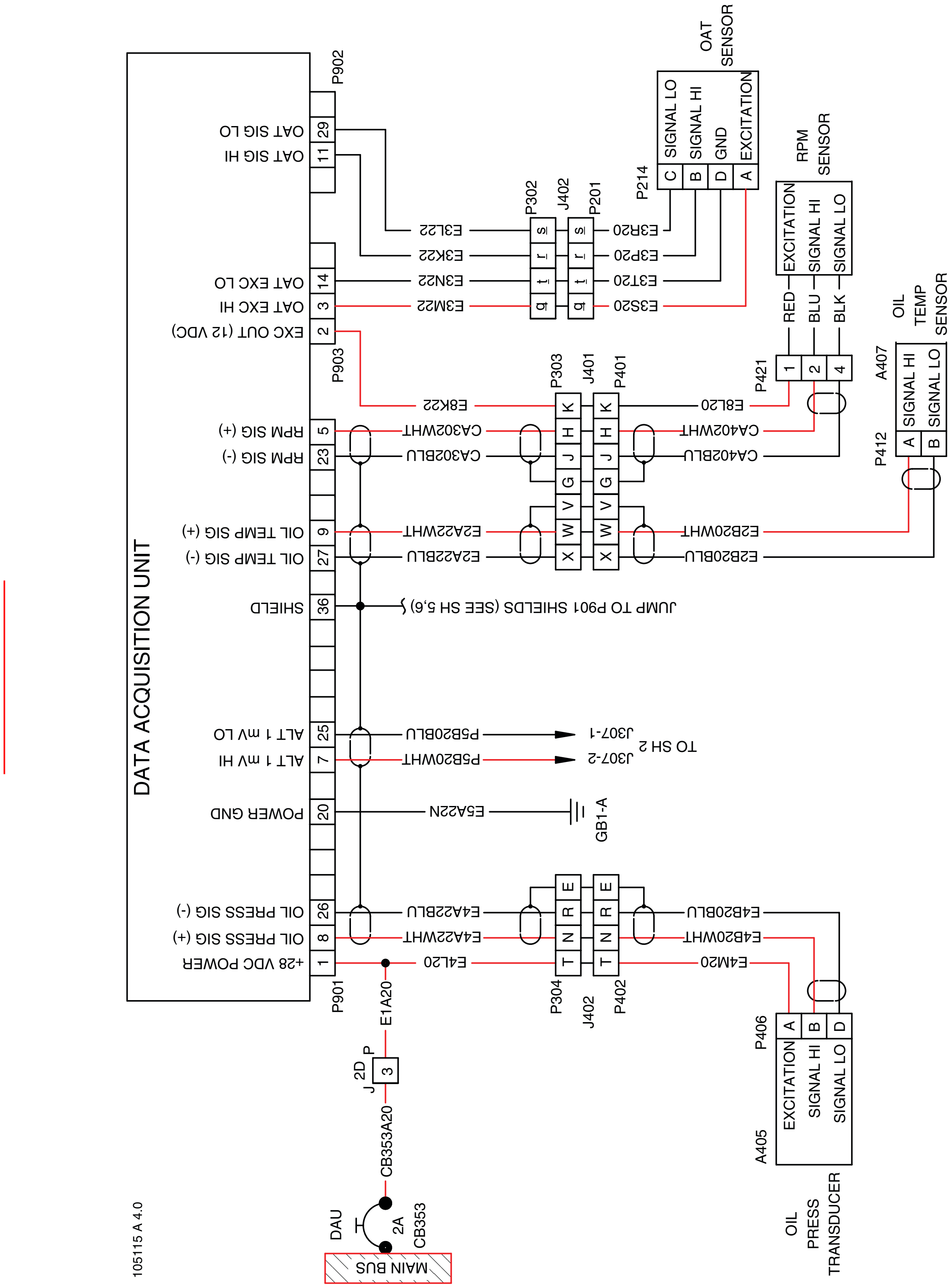


Enhanced Digital Indicator (EDI)
Figure 2 (Sheet 6 of 6)

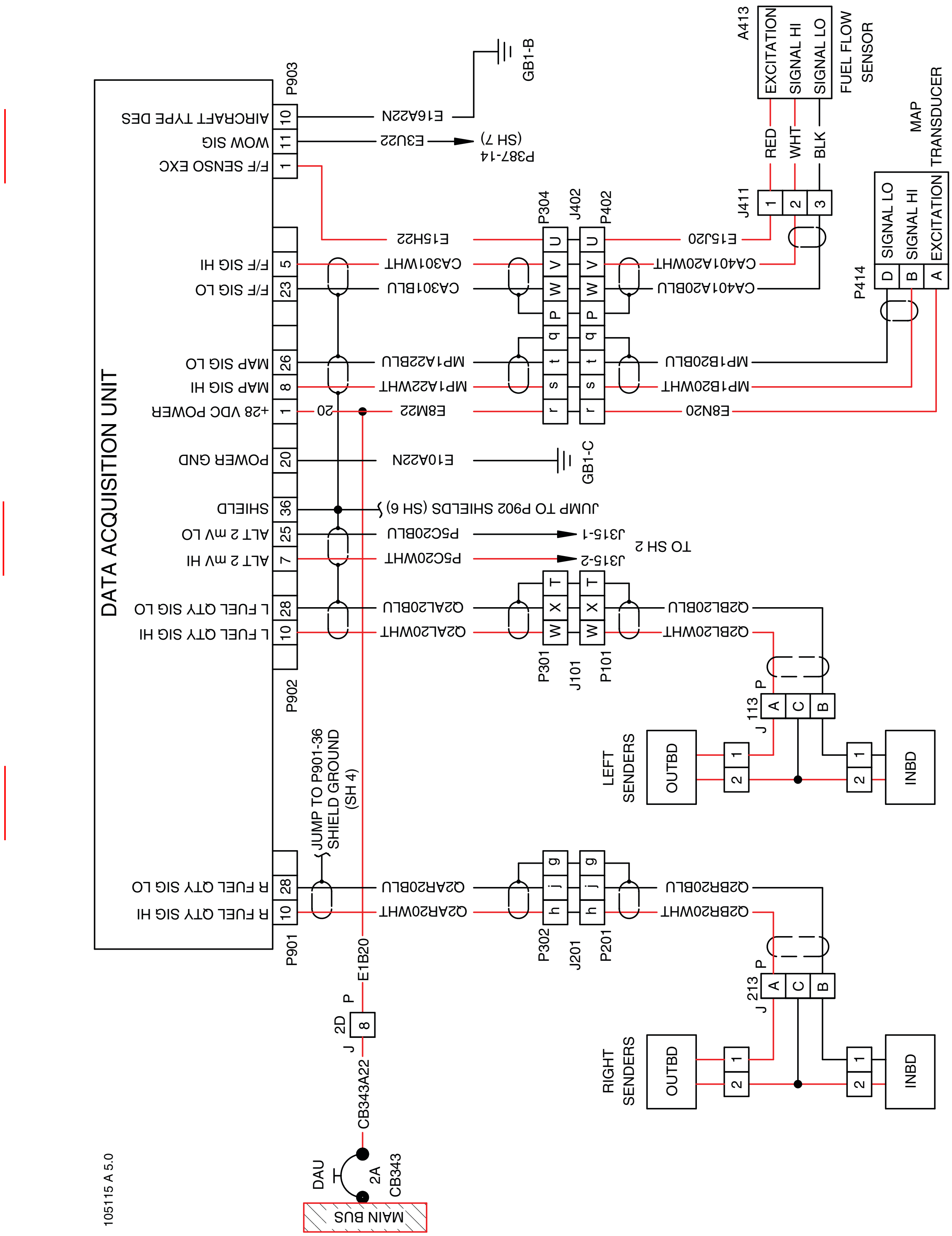
Item #	Designation	Description
1		Data Acquisition Unit (DAU)
2		RPM Sensor (on Left Magneto)
3	A405	Oil Pressure Transducer
4	A407	Oil Temp Sensor
5		OAT Sensor
6		Fuel Senders (Left & Right; Inboard & Outboard)
7		MAP Transducer
8	A413	Fuel Flow Transducer
9	A441 thru A446	CHT Sensors
10		TIT Sensor
11	A307	Vacuum Transducer
12	CB343, CB353. CB7*	Circuit Breaker (2 Amp)
13	CB313, CB14*	Circuit Breaker (3 Amp)



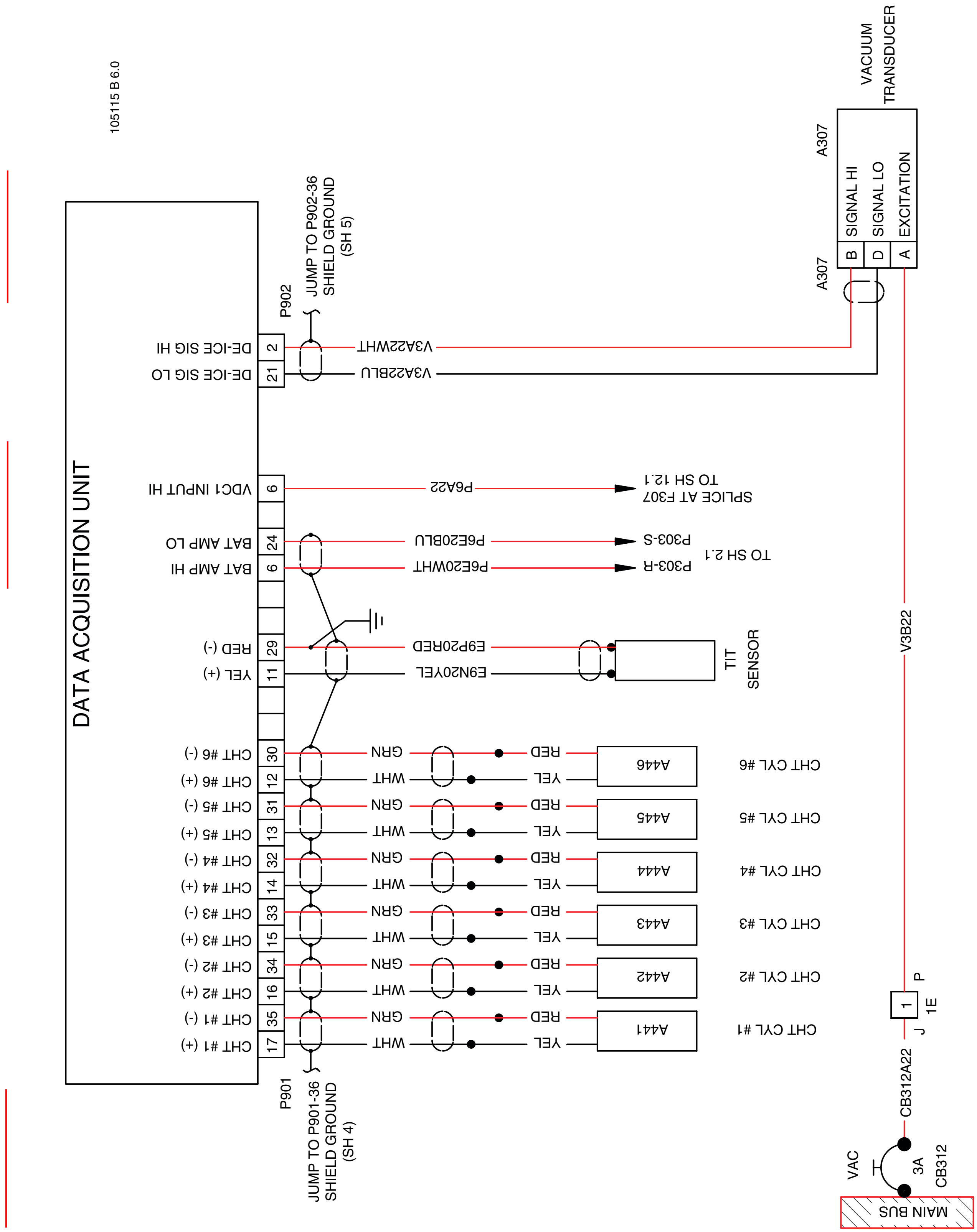
Data Acquisition Unit (DAU)
Figure 3 (Sheet 1 of 7)



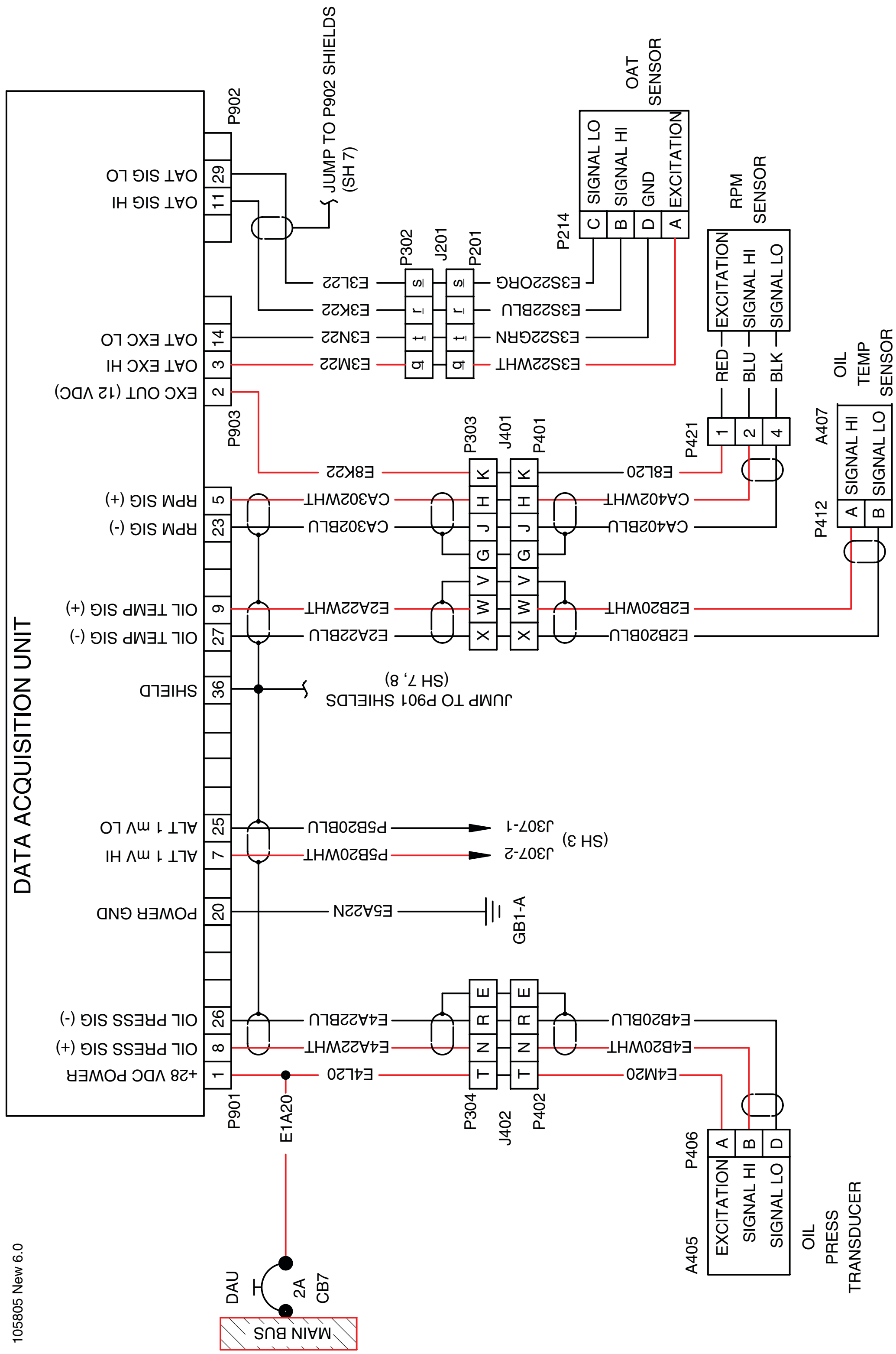
Data Acquisition Unit (DAU)
Figure 3 (Sheet 2 of 7)



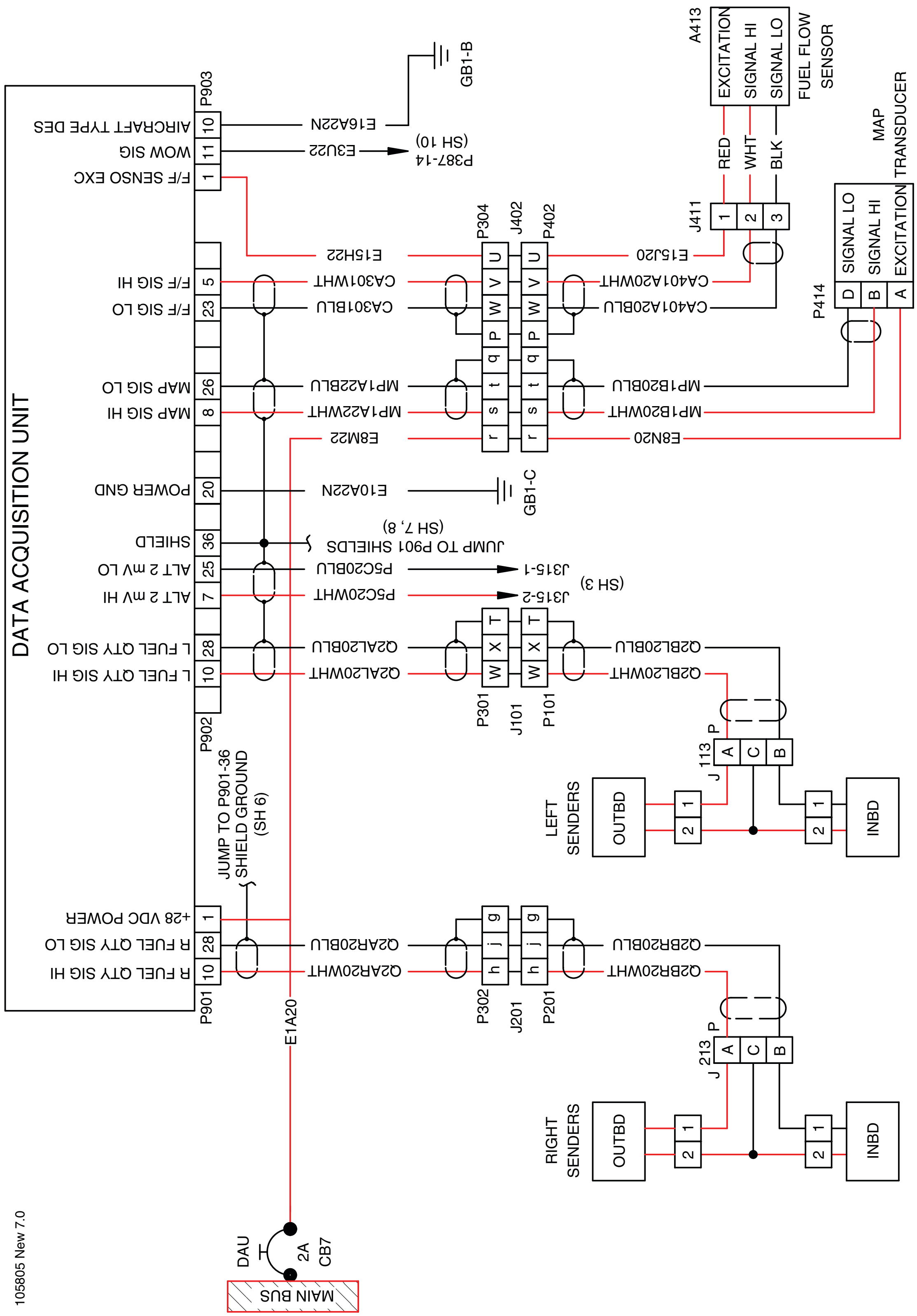
Data Acquisition Unit (DAU)
Figure 3 (Sheet 3 of 7)



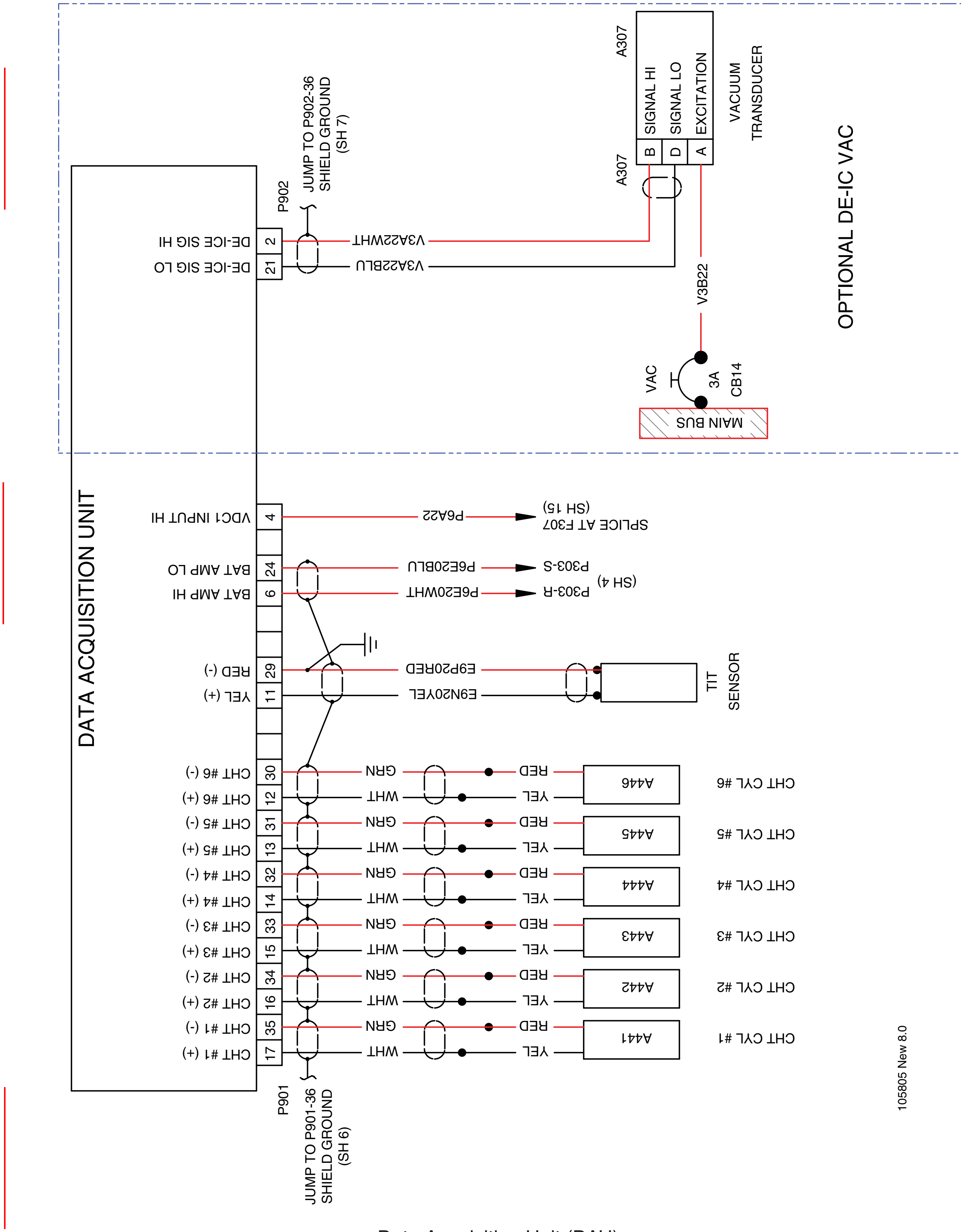
Data Acquisition Unit (DAU)
Figure 3 (Sheet 4 of 7)



Data Acquisition Unit (DAU)
Figure 3 (Sheet 5 of 7)



Data Acquisition Unit (DAU)
Figure 3 (Sheet 6 of 7)



Data Acquisition Unit (DAU)
Figure 3 (Sheet 7 of 7)

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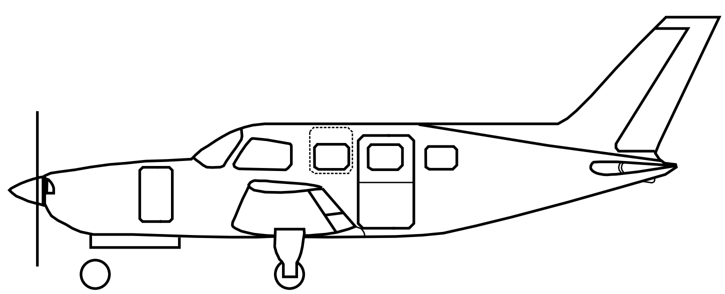
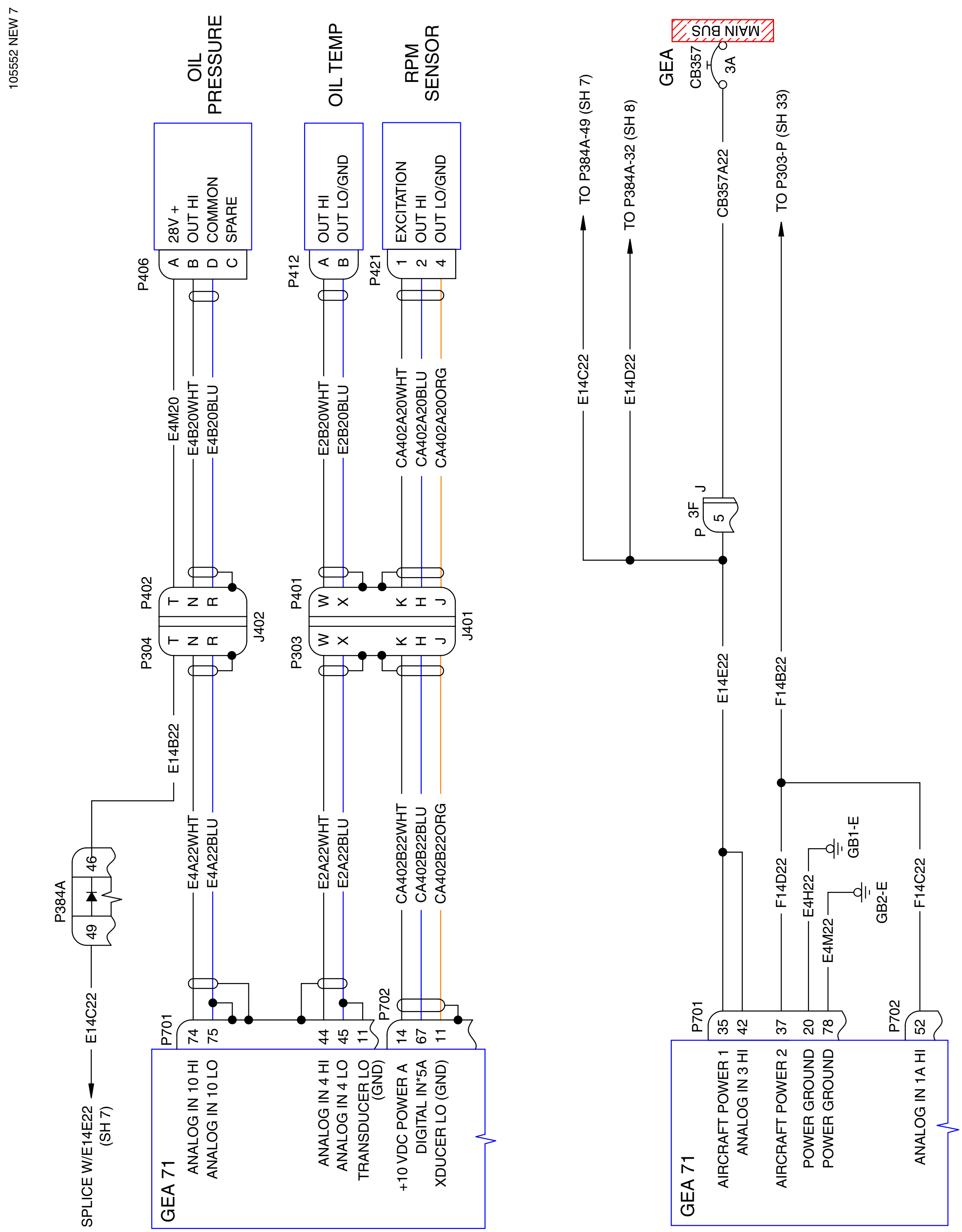
Item #	Designation	Description

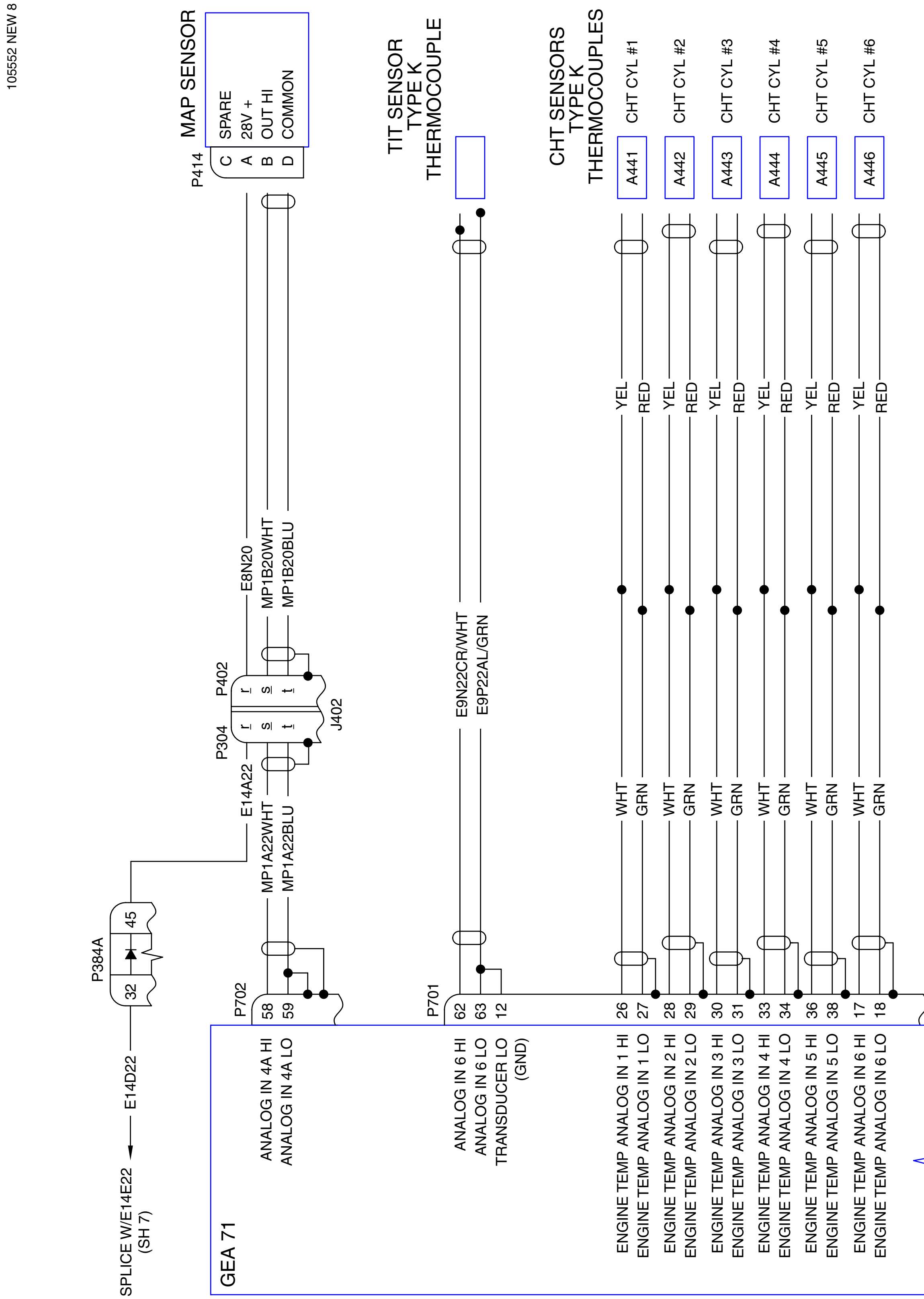
Information Pending - See Parts Catalog



IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 1 of 9)



IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 2 of 9)



IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 3 of 9)

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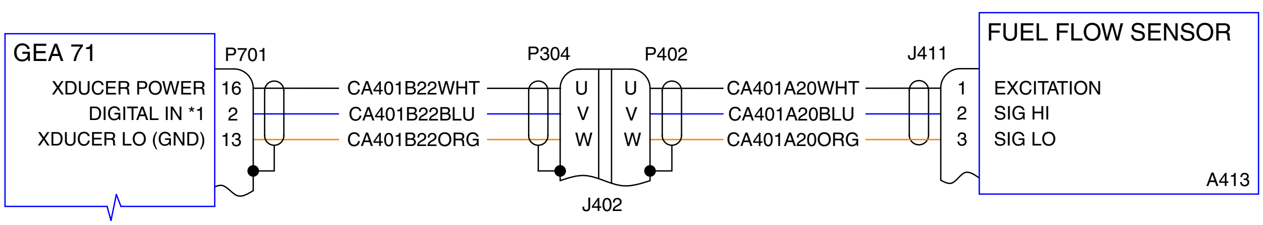
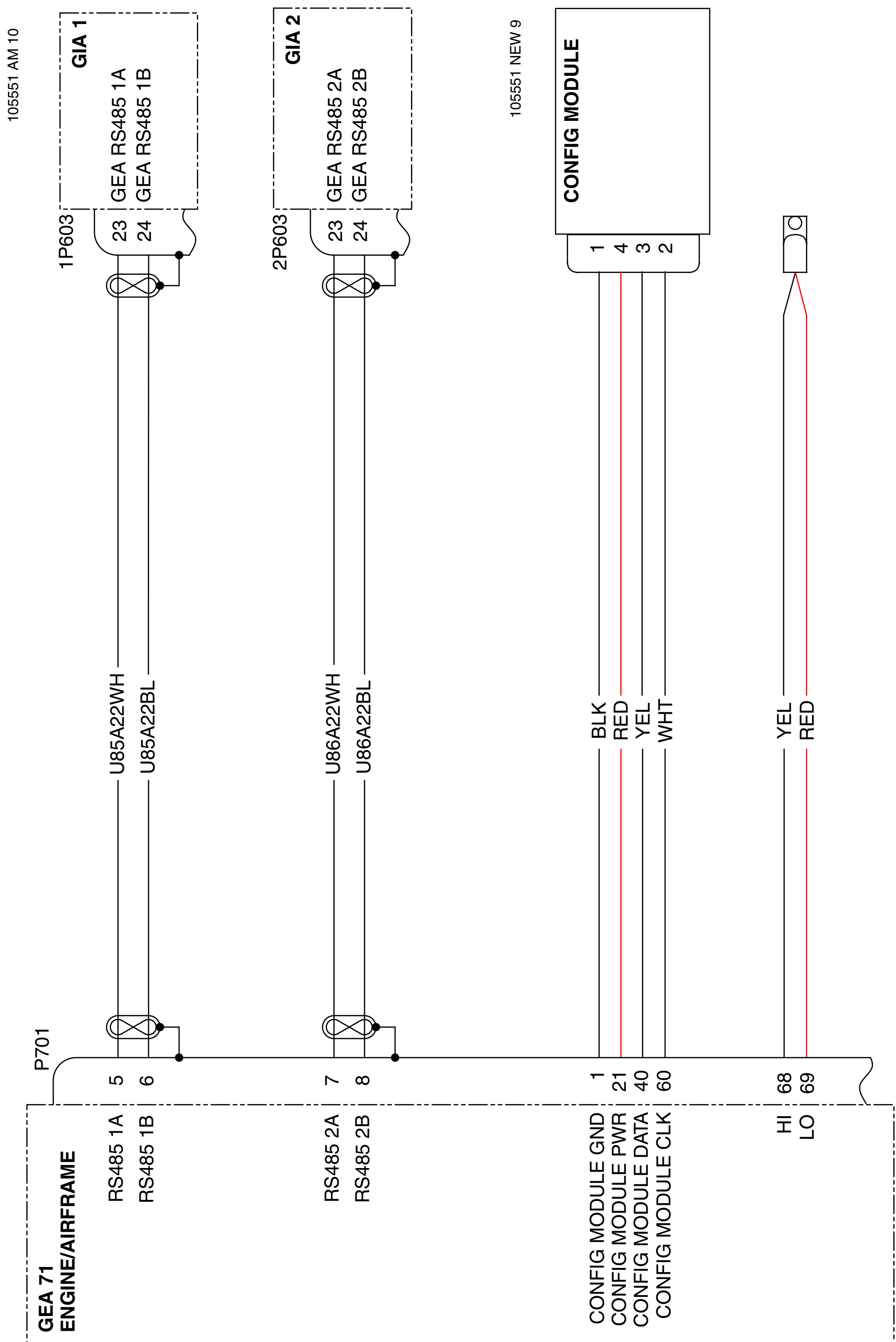
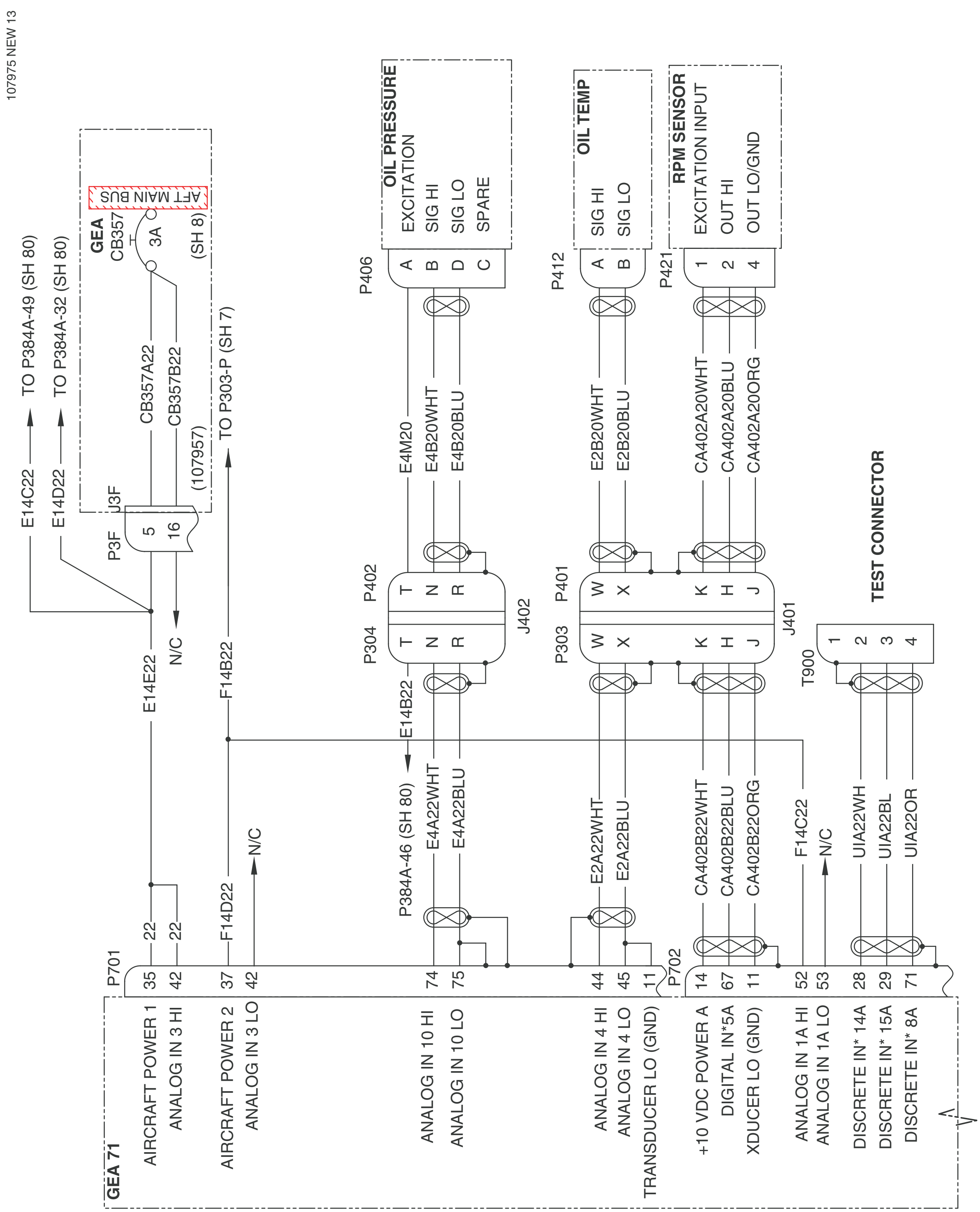


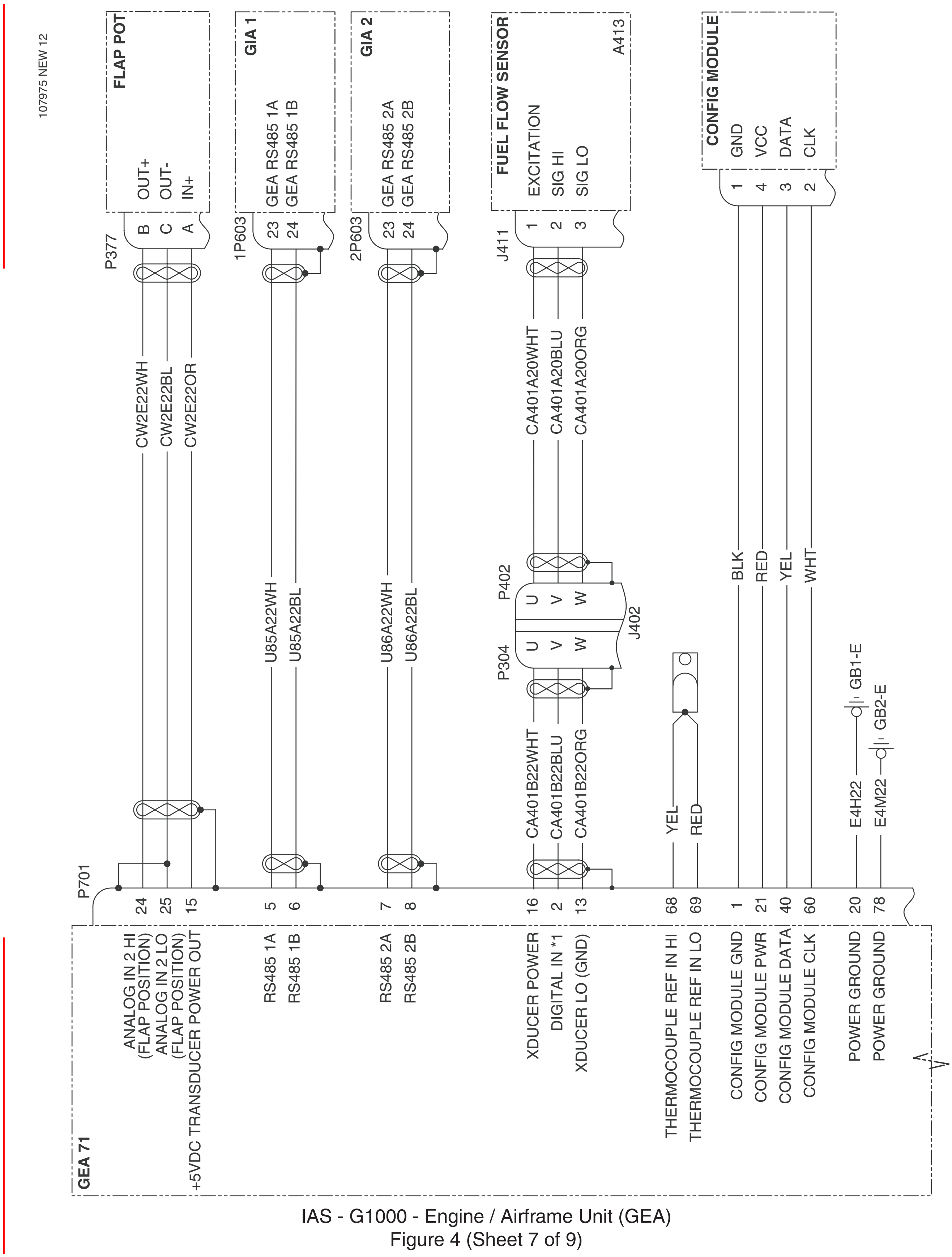
IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 4 of 9)



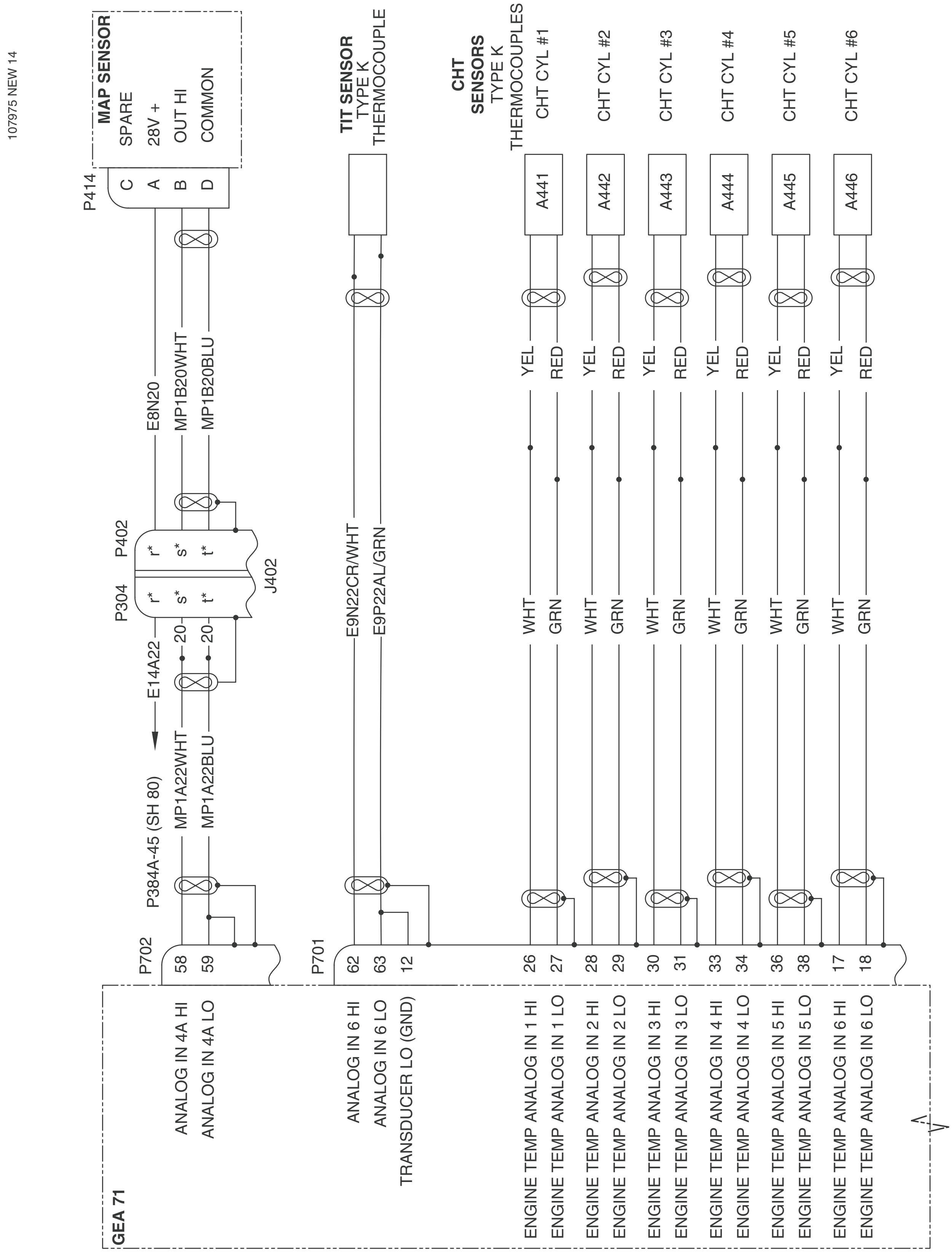
IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 5 of 9)



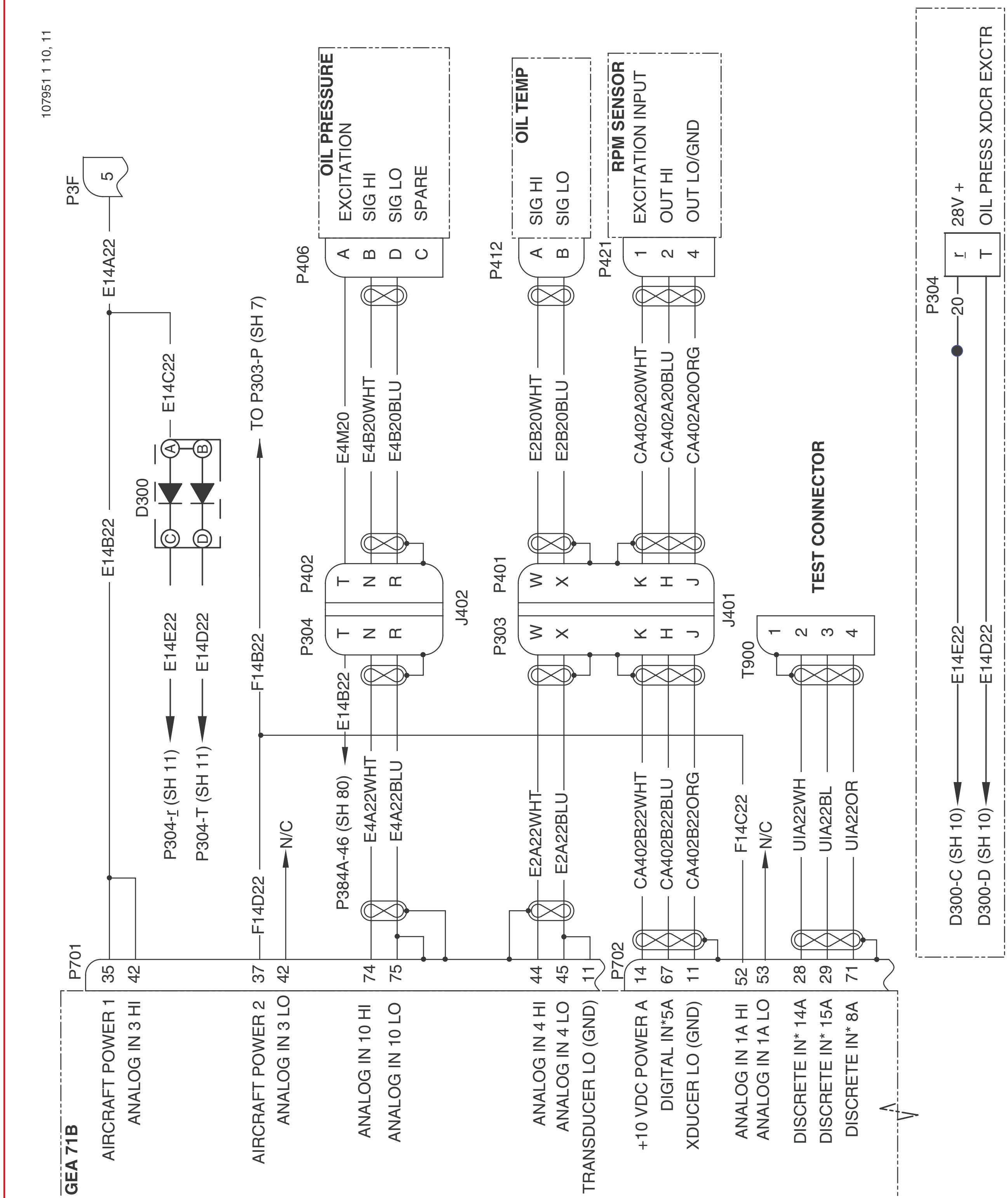
IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 6 of 9)



IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 7 of 9)



IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 8 of 9)



IAS - G1000 - Engine / Airframe Unit (GEA)
Figure 4 (Sheet 9 of 9)

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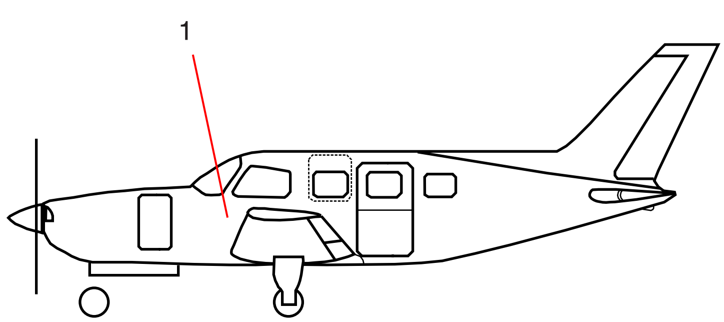
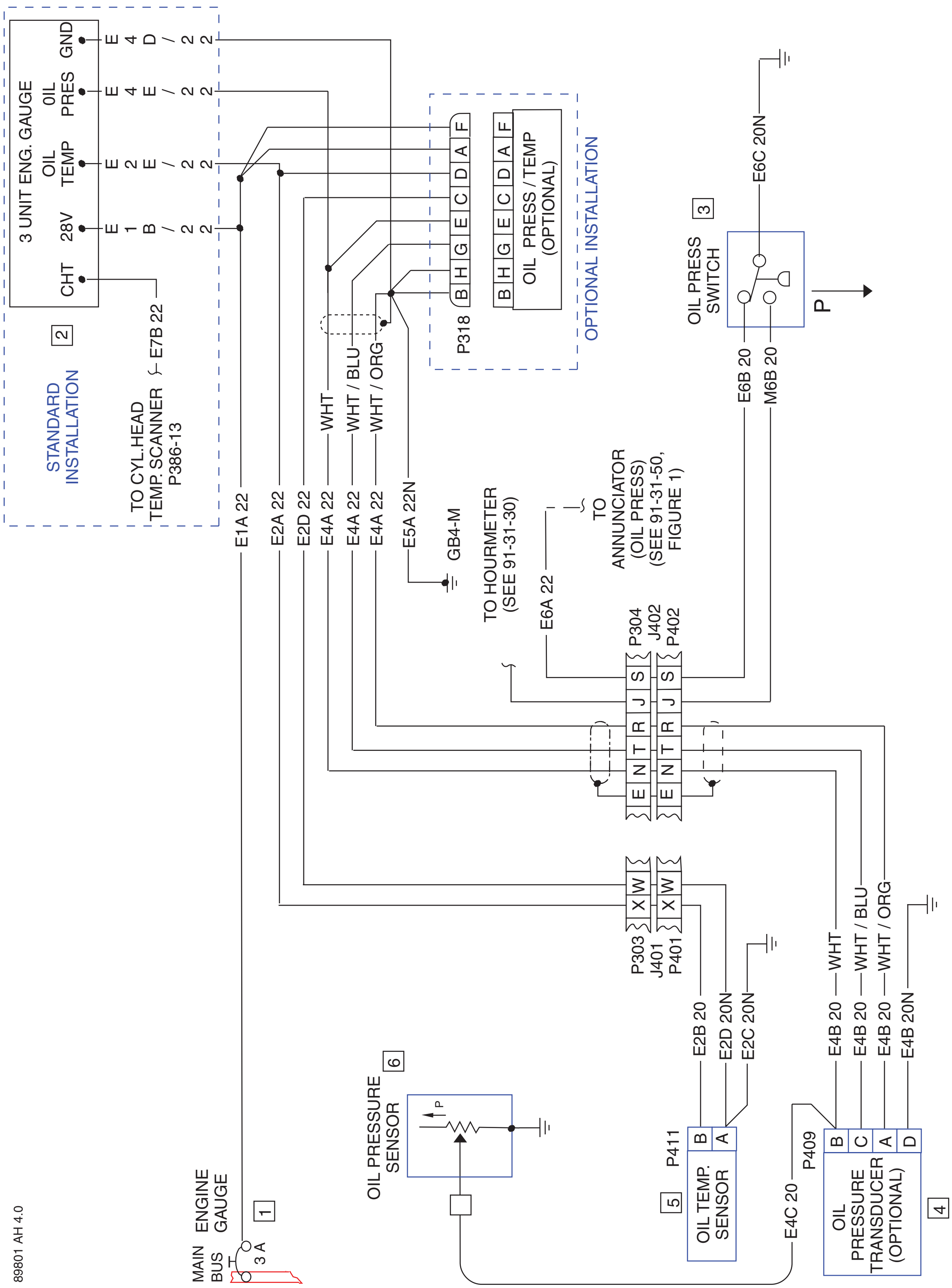
Item #	Designation	Description
1	CB315	Circuit Breaker (3 Amp)
2	M30	3-Unit Engine Gauge
3	A404	Oil Pressure Switch
4	A405	Oil Pressure Transducer
5	A407	Oil Temperature Sensor
6		Oil Pressure Sensor



Oil Pressure and Temperature Indicators
Figure 1 (Sheet 1 of 2)



Oil Pressure and Temperature Indicators
Figure 1 (Sheet 2 of 2)

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APPENDIX

1

GROUND TEST PROCEDURE

MEGGITT MAGIC AIR DATA & HEADING REFERENCE SYSTEM (ADAHRS)
AND
PITOT-STATIC SYSTEM

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APPENDIX 1

LIST OF EFFECTIVE PAGES

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1. Reference Regulations and Guidance

Title 14 of the Code of Federal Regulations (14 CFR), Part 43 Appendix E, 23.1301(d), 23.1323(a)(c), 23.1325(b), 91.411, Advisory Circular AC 43-6B, and Advisory Circular AC 43.13-1B CHG 1 (or later) Chapter 12, Section 4.

2. Reference Documents

- Applicable Airplane Flight Manual (AFM)
- Applicable Airplane Maintenance Manual
- Applicable Meggitt Avionics/S-TEC MAGIC Pilot's Operating Handbook (POH)
- Meggitt Avionics/S-TEC Airspeed Configuration Data document (if applicable) for the airplane under test.

3. System Description

Each Meggitt MAGIC Electronic Flight Instrument System (EFIS) consists of the following components related to the Air Data and Pitot-Static systems:

- Primary Flight Display (PFD) with primary altimeter
- Navigation Display (ND) with reversionary altimeter
- Air Data/Attitude and Heading Reference System (ADAHRS)
- Standby Altimeter
- Standby Airspeed Indicator
- ATC Transponder (if installed)

Refer to the appropriate installation drawings or Instructions for Continued Airworthiness (ICA) document to determine the specific part numbers and locations of these components.

4. General

Purpose. This test procedure verifies the integrity and accuracy of the air data and pitot-static systems on the aircraft under test and shows that these systems comply with applicable airworthiness regulations. No bench testing or field repairs are included in this procedure. Any unsatisfactory findings should be repaired in accordance with other acceptable means.

Use this on-aircraft ground test procedure (GTP) to inspect and test each Meggitt Avionics/S-TEC MAGIC Air Data and Pitot-Static system, under the following conditions:

- In accordance with 24-month static system test; or
- To proof test static systems following any opening and closing of the pitot-static plumbing, except for the use of system drain and alternate static pressure valves (§ 23.1325(b)(2); or
- Following installation or maintenance on Air Data/Attitude and Heading Reference System (ADAHRS), Primary Flight Display (PFD), or Transponder Mode C interface.

5. Effectivity

This GTP is effective to aircraft with Meggitt Avionics/S-TEC MAGIC EFIS and pneumatic standby instruments.

6. Prior to testing

Conduct this GTP after the following has been accomplished:

- Continuity testing of new or altered electrical wiring
- Power and ground testing of new or altered electrical wiring
- Electrical bonding testing of new equipment
- Installation of equipment

Ground Test Procedure (Sheet 1 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

- Required test equipment, technical data, and rated personnel are available to perform a static system check as required by § 91.411(a)(2).
- Calibrated pitot-static tester and transponder ramp tester is available to determine the pressure altitude transmitted by the transponder in accordance with § 91.217(b).
- Completion of appropriate EFIS Post-Installation GTP.

NOTE
This Pitot-Static GTP is to be accomplished after MAGIC EFIS Post-Installation GTP and after leveling of the ADAHRS, which may require that the pitot-static lines to be temporarily disconnected.

7. After testing

Ensure that an appropriately rated person returns the aircraft to service following the alteration or maintenance in accordance with § 43.9.

8. Standard atmospheric ambient conditions and tolerances

Unless otherwise specified, testing shall be performed during the following standard atmospheric ambient conditions:

- Temperature: 25° C ± 10° C
  Record temperature if outside of range: _________

Avoid calibrating air data systems at extreme temperatures.

9. Precautions

This section contains precautions that should be taken during tests to avoid damage to the sensitive instruments.

Damage can occur to instruments that are connected to both the static system and pitot system when only the static system is evacuated. The maximum design differential pressure of these instruments may be exceeded. This GTP requires that both the pitot and static ports be connected to the pitot-static tester.

Safeguards should be taken to prevent accidental disconnection of the test equipment plumbing from the aircraft or the test equipment while the static system is evacuated. The resultant sudden pressure change may damage both the test instruments and the aircraft instruments. The aircraft static system should be returned to ambient pressures before disconnecting static test equipment from the system

If static ports are blocked for the purpose of testing, use a long piece of brightly colored tape (red or orange) forming a streamer or similar method of warning be attached to the blockage device.

Damage may occur to other aircraft instruments if the altitude rate is changed faster than the limit of instruments.

10. Calibration of equipment

Conformity aircraft. Unless otherwise specified, test instruments used to calibrate or verify calibration in a certification project should be calibrated by an approved instrument repair inspector prior to FAA testing. Test instruments should be calibrated within 3 months of FAA testing. Reference AC 21-40, Section 8-4(h).

Follow-on installations and maintenance. In accordance with § 145.47 (b), the repair station shall ensure that all inspection and test equipment is tested at regular intervals to ensure correct calibration to a standard derived from the National Bureau of Standards or to a standard provided by the equipment manufacturer.

Ground Test Procedure (Sheet 2 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

11. Ground Equipment

Use the following ground equipment for this GTP.

1. Ground Power Unit (GPU). Conduct this test procedure using a GPU that will provide sufficient regulated electrical power to the aircraft.
2. Pitot-Static tester. Use a calibrated pitot-static tester to simulate airspeed and altitudes on the aircraft. It is recommended to use a tester with a remote electronic control panel to facilitate the verification of the instruments in the aircraft.
3. ATC Transponder Ramp Tester. Use a calibrated ATC Transponder ramp tester to interrogate and verify proper Mode C (altitude) altitude reporting.
4. Barometer (pressurized aircraft only). A calibrated barometer may be used to determine the "station pressure" at the field where the aircraft will be tested. Station pressure will be used to calculate the MAX DIFF test altitude, which is the test altitude where the pitot-static system will be tested for leaks. (*Do not use altimeter setting barometric pressure for the MAX DIFF calculation.*)

CAUTION
Remove all tape and thoroughly clean ports to remove any adhesive resins after testing.

12. Ground Equipment ID

Record in Table I below the specific ground equipment used during this GTP:

Table I — Test Equipment

TEST EQUIPMENT	MANUFACTURER	MODEL	S/N	CAL DATE	CAL DUE
Ground Power Unit (GPU)					
Pitot-Static Tester					
ATC Transponder Tester					
Barometer					

13. Local information

Field elevation. Record the field elevation (H_a) where the aircraft is being tested, as a reference.

Field Elevation (H_a)	feet

14. Proof Test Parameters (Pressurized Aircraft Only)

MAX DIFF test altitude (computation). For pressurized aircraft only, determine the equivalent altitude of the maximum cabin pressure differential (MAX DIFF test altitude).

This is the test point at which the proof test will be accomplished. The test altitude will vary depending on the current station barometer and the cabin pressurization limit of the aircraft under test. The following steps are provided to compute the MAX DIFF test altitude:

Step 1 (Station Pressure): Obtain the current local barometric pressure (station pressure) within 25 miles. (*Note that the station barometer is the true barometric pressure, unlike altimeters settings, which are "reduced" to sea level pressure*):

Station Barometer (Inches of Mercury)	In Hg

Step 2 (Barometer Conversion): Convert the station barometric pressure to PSI, by dividing the barometric pressure by 2.036:

Station Barometer (Pounds per Square Inch)	PSI

Step 3: (Cabin Pressurization Limit). Obtain the approved maximum cabin differential pressure for the aircraft under test, which is usually found in the AFM Limitations:

Maximum Cabin Differential Pressure	5.5 PSI

Step 4 (MAX DIFF test pressure). Subtract Step 3 from Step 2 to determine the MAX DIFF test pressure:

MAX DIFF test pressure	PSI

Step 5 (MAX DIFF test altitude). Use Table IX at the end of this document to approximate the test pressure of Step 4 to an equivalent altitude, by selecting the next lower number in the PSI column and recording its corresponding altitude below:

Equivalent Altitude of Maximum Cabin Pressure Differential (MAX DIFF test altitude)	feet

EXAMPLE:

Step 1. Local station pressure: 25.39 Inches of Mercury

Step 2. $PSI = \frac{25.39}{2.036} = 12.47$ PSI

Step 3: Cabin pressurization limit: 5.3 PSI

Step 4: 12.47 – 5.3 = 7.17 PSI

Step 5: 7.17 PSI ≈ 19000 feet

The MAX DIFF test altitude of 19000 feet was estimated by rounding 7.17 PSI to the next lower value of 7.0412, as shown in Table IX at the end of this document.

MAX DIFF test altitude (**alternate method**). An alternate method to determine MAX DIFF test altitude on aircraft with an approved maximum differential cabin pressure indicator is to use the redline mark on this indicator. By this method, use the pitot-static tester to increase the altitude of the static system until the cabin differential pressure indicator reaches redline. Record the corresponding altitude as the MAX DIFF test altitude.

Ground Test Procedure (Sheet 4 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

MAX DIFF tolerance. Calculate the MAX DIFF tolerance of 2% of MAX DIFF test altitude. Multiply MAX DIFF test altitude by **0.02** to obtain the tolerance. Insert this value into Table IV also.

MAX DIFF Tolerance = (0.02) x (MAX DIFF): | ± feet |

15. Airspeed and Altitude Limits

Record the airspeed and altitude design limits of the aircraft under test, as specified in the appropriate Airplane Flight Manual (AFM) or Type Certificate Data Sheet (TCDS). Use these figures to determine the limits of testing. Do not exceed the airspeed limit by more than 10 knots or the altitude limit by more than 2500 feet.

Airspeed limit. Record the maximum operating airspeed of the aircraft under test. Also make a note of the maximum airspeed in the margin of Table VIII for the Indicated Airspeed Test.

Maximum Operating Airspeed: | * knots |

Altitude limits. Record the maximum operating altitude of the aircraft under test. Also make a note of the maximum altitude in the margin of Table V for the Scale Error Test and Table VII for the Friction Test.

Maximum Operating Altitude: | ** feet |

16. Hysteresis Test

Dive (hysteresis) test points. Calculate required hysteresis test points from the maximum operating altitude.

50% test point. Multiply the maximum operating altitude by **0.5**. From this value, select the closest test point from the altitudes listed in Table V as the 50% test point:

50 % test point: | feet |

40% test point. Multiply the maximum operating altitude by **0.4**. From this value, select the closest test point from the altitudes listed in Table V as the 40% test point:

40 % test point: | feet |

Record the 50% and 40% test points in the "*Calculated Test Point Altitude*" column of Table VI.

* 188 KIAS - PA-46-500TP
198 KIAS - PA-46-350P

** 30,000 FT - PA-46-500TP
25,000 FT - PA-46-350P

Ground Test Procedure (Sheet 5 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
1.	**Before Applying Electrical Power**				
2.	GROUND SAFETY INSPECTION Conduct a ground safety inspection in accordance with the appropriate Airplane Maintenance Manual prior to applying electrical power to the aircraft under test.	Check that master switches are OFF. Pull circuit breakers for engine start and fire control OUT.			
3.	Remove pitot-static covers (if installed)	Check that pitot-static ports are clear.			
4.	**Inspection**				
5.	*The following inspections are in compliance with § 23.1325(b)(1) and § 43 Appendix E(a).*				
6.	*<u>Note</u>: Whenever blockage of the static lines is suspected, they should be purged before the static pressure system test is performed. Purging may keep objects from entering the test equipment and instruments. Since purging applies positive pressure to lines, the following precautions should be taken:* *(1) Disconnect all instruments, sensors, and ADAHRS from the pitot-static lines.* *(2) Secure a cloth bag over the end of the lines to catch any debris.* *(3) Clean drains after purging as they can act as a sump for foreign material.*				
7.	Inspect and operate the pitot-static drains.	Verify drains operate normally and that positive drainage of moisture is provided in all pitot-static lines.			
8.	Inspect installation of the pitot-static lines.	Verify that chafing of the tubing and excessive distortion or restriction at bends in the tubing is avoided.			

Ground Test Procedure (Sheet 6 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
9.	Inspect the ports, tubing, accessories, and instruments connected to the static system to identify any parts that might be defective (e.g., broken "B" nuts, cracked flare sleeves, deteriorated flexible tubing, bad valves, etc.).	Verify that the materials used are durable, suitable for the purpose intended, and protected against corrosion.			
10.	Inspect static ports.	Verify static ports are free of debris and corrosion.			
		Verify that no alterations or deformations of the airframe surface have been made in the areas near the static ports.			
11.	Inspect alternate static ports.	Verify alternate static port is free of debris or corrosion.			
12.	Inspect pitot probes.	Verify inlet is free of debris or damage.			
		Verify that the pitot probe does not have any corrosion within ½-inch of probe tip.			
		Ensure that no alterations or deformations of the airframe surface have been made in the areas near the pitot probes.			
13.	Inspect transponder antennas. *Note: Oil, exhaust, and dirt buildup on a transponder antenna may attenuate and shift frequency out of tolerance.*	Verify transponder antennas are clean and not damaged.			
14.	**Set up ground equipment**				
15.	Set up Transponder ramp tester with the test antenna near the transponder antenna.				
16.	Connect Ground Power Unit to aircraft.				

Ground Test Procedure (Sheet 7 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
17.	**Apply electrical power to aircraft**				
18.	Activate ground power in accordance with the Airplane Maintenance Manual procedures and turn on electrical buses.				
19.	Turn on cabin lights to verify electrical power is applied to the aircraft.				
20.	Turn on EFIS power. *Check that PFD is not in reversionary mode.*	Verify normal PFD screen after PFD initialization screen. Verify ADAHRS initializes after normal countdown.			
21.	Turn on static port heat (if installed).	Verify either increased ammeter current or that static port becomes hot to the touch.			
22.	Turn off static port heat (if installed).	Verify either reduced ammeter current or that static port returns to ambient.			
23.	Turn on pitot heat.	Verify either increased ammeter current or that pitot probe becomes hot to the touch.			
24.	Turn off pitot heat.	Verify either reduced ammeter current or that pitot probe returns to ambient.			

Ground Test Procedure (Sheet 8 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
25.	**Alternate Static (if installed) Test at Field Elevation**				
26.	***Purpose**. This test will verify that the altimeters do not change more than 50 feet between normal and alternate static sources at field elevation, in compliance with § 23.1325(b)(3).* ✈ *A functional flight check of the alternate static selector is recommended.*				
27.	Press **BARO** knob on PFD 1 and 2.	Verify PFD indicates STD.			
28.	Set BARO on standby altimeter to **29.92**.	Verify dial rotates normally.			
29.	Inspect the alternate static source selector.	Verify alternate static selector is clearly marked and can be switched freely between normal and alternate sources.			
30.	Switch alternate static to NORMAL.	Record results in the NORMAL row in Table II below.			
31.	Switch alternate static to ALTERNATE.	Record results in the Alternate row in Table II below.			
32.	Return alternate static to NORMAL.	Compute the difference between normal and alternate in Table II below. Verify that ~~the altitude~~ does not differ by more than 50 feet, IAW § 23.1325(b)(3).			

Table II — Alternate Static Test

Alternate Static Switch	Pilot's Altimeter (feet)	Copilot's Altimeter (feet)	Standby Altimeter (feet)	Tolerance (feet)	Pass/Fail
Normal					
Alternate					
Difference				±50	

Ground Test Procedure (Sheet 9 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
33.	**Barometric Scale Error Test**				
34.	***Purpose***. *This test verifies the accuracy of the barometric scale on each altimeter, in compliance with § 43 Appendix E, Table IV.*				
35.	Set the BARO knob on each altimeter to standard pressure 29.92 in order to determine the pressure altitude (Pa) at field elevation.	Record the pressure altitude (Pa) in the first row of Table III below. Use this altitude as the reference point.			
36.	Adjust the BARO knob on each altimeter to the specified settings, record the new indicated altitudes, and compute the difference between the indicated altitudes from pressure altitude.	Verify that the change in altitude on each altimeter is within ±25 feet of the expected change from pressure altitude at field elevation.			

Table III — Pressure-Altitude Difference

BARO Setting	Expected Change from Pressure Altitude (Pa)	Tolerance	PFD 1 Altimeter (feet)		PFD 2 Altimeter (feet)		Standby Altimeter (feet)		Pass/Fail
			Indication	Altitude change from Pa	Indication	Altitude change from Pa	Indication	Altitude change from Pa	
29.92	Pressure Altitude (PA)								
29.50	-392	±25 ft							
29.00	-863	±25 ft							
28.50	-1,340	±25 ft							
28.10	-1,727	±25 ft							
30.50	+531	±25 ft							
30.90	+893	±25 ft							
30.99	+974	±25 ft							

37.	Press **BARO** knob on PFD 1 and 2.	Verify PFD indicates STD.			
38.	Reset BARO on standby altimeter to **29.92**.				

Ground Test Procedure (Sheet 10 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
39.	**Pitot-Static Tester Set Up**				
	Test Points. *Connect the pitot-static test equipment directly to the pitot-static ports on the aircraft, if practicable. Otherwise, connect to a static system drain or tee connection and seal off the static ports. If the test equipment is connected to the static system at any point other than the static port, it should be made at a point where the connection may be readily inspected for system integrity after the system is returned to its normal configuration.* ***Seals.*** *Block unconnected ports with brightly colored tape long enough to have a streamer.* **Note:** Remove all static port seals after completion of this test procedure.				
40.	***Separation.*** *If the aircraft under test has more than one pitot-static system, test each system separately to ensure their independence and that the leak rate for each is within tolerance.* ***Vibration.*** *Unless otherwise specified, each test may be conducted with vibration applied to pneumatic standby instruments.* **Note:** **When applying vibration, lightly tapping the standby instrument face with a finger should suffice.** **DO NOT install a shaker or vibrator on the instrument panel.** **Caution:** Secure pitot-static test connections carefully. Avoid circumstances that would result in a test line disconnecting unintentionally while test pressure is being applied.				

Ground Test Procedure (Sheet 11 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
41.	**Proof Test**				
	Purpose. *This test will verify the integrity of the pitot-static system(s), including the lines and components, in compliance with § 23.1325(b)(2) and § 91.411(a)(2). Procedures are included for both pressurized and non-pressurized aircraft, with the allowable tolerance of each type shown. Use one or the other as appropriate for the aircraft under test.* ***Method.*** *Each pitot-static system shall be tested separately to verify that it can hold the test pressure for 1 minute without support from the vacuum pump in the tester. The pitot and static ports of each system shall be connected to the pitot-static tester to verify both lines.*				
42.	**Note:** On aircraft with multiple pitot systems using a single static system, connect all pitot systems to the pitot-static tester to avoid damage to instruments. **Caution:** Monitor all air data instruments during testing to ensure that none are operated beyond limitations. *All instruments in the aircraft under test shall be connected to their normal lines to assure that no leaks occur within the instruments or their connections.* *The primary altimeter in the aircraft under test or that in the pitot-static tester may be used as a vacuum gauge for the proof test.* *Use Table IV to record readings, calculations, and pass/fail results.*				

Ground Test Procedure (Sheet 12 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
43.	Unpressurized Airplanes. With the pitot-static tester connected to each pitot-static system separately, simulate a test altitude of 1,000 feet above field elevation at 0 knots airspeed. Upon reaching the test altitude, record the indicated altitude displayed on the specified altimeter as the *Initial Altitude* in Table IV. *After 1 minute* without additional vacuum being applied to the aircraft, record the second altitude from the same altimeter.	Compute the *Altitude Loss* in Table IV by subtracting the *Initial Altitude* from the *Altitude After 1 Minute.* Verify that the *Altitude Loss* is less than the unpressurized proof test tolerance of ±100 feet.			
44.	Pressurized Airplanes. With the pitot-static tester connected to each system separately, simulate MAX DIFF test altitude, as computed in Section 14: ▶ ________________ feet Upon reaching the MAX DIFF altitude, record the indicated altitude displayed on the specified altimeter as the *Initial Altitude* in Table IV. *After 1 minute* without additional vacuum being applied to the aircraft, record the second altitude from the same altimeter.	Compute the *Altitude Loss* in Table IV by subtracting the *Initial Altitude* from the *Altitude After 1 Minute.* Verify that the *Altitude Loss* is less than the Pressurized MAX DIFF Tolerance (2% of MAX DIFF altitude).			
45.	Return pitot-static tester to atmospheric conditions on conclusion of proof test.				

Table IV — Proof Test

Parameter		Pilot's Altimeter (feet)	Copilot's Altimeter (feet)	Standby Altimeter (feet)	Unpressurized Proof Test Tolerance(feet)	Pressurized MAX DIFF Tolerance(feet)	Pass/Fail
1st reading →	*Initial Altitude*						
2nd reading →	*Altitude After 1 Minute*						
Calculate →	*1st – 2nd = Altitude Loss*				±100	▶ ±	

Ground Test Procedure (Sheet 13 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
46.	**Scale Error and Altitude Reporting Test**				
47.	***Purpose.*** *This test will verify the accuracy of the altimeters and altitude-reporting devices during a 4000--6000 FPM climb between test points, in compliance with Part 43 Appendix E, Table I and § 91.217(b).* ***Method.*** *Each static system may be tested separately or together. The pitot and static ports of each system shall be connected to the pitot-static tester.* **Caution:** Monitor all air data instruments during testing to ensure that none are operated beyond limitations. *All instruments in the aircraft under test shall be connected to their normal lines.* *Each altimeter in the aircraft under test will be compared to the altitude of the tester at test point altitudes, as specified in Table V.* *Use Table V to record readings, calculations, and pass/fail results.*				
48.	Set pitot-static tester altimeter to 29.92 and airspeed to 0.				
49.	Press the BARO knob on PFD 1 and 2 for STD (29.92) setting.				
50.	Set the BARO on standby altimeter to 29.92.				
51.	Set up the transponder ramp tester to interrogate the transponder(s) on the aircraft.				
52.	Turn on transponder 1 in Mode 3/A with altitude (Mode C) off.	Verify on the ramp tester that the transponder is NOT transmitting altitude information.			
53.	Turn on altitude (Mode C) on transponder 1.	Verify proper altitude pulse train or decoded altitude on the ramp tester.			

Ground Test Procedure (Sheet 14 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
54.	Turn on transponder 2 (if installed) in Mode 3/A with altitude (Mode C) off.	Verify on the ramp tester that the transponder is NOT transmitting altitude information.			
55.	Turn on altitude (Mode C) on transponder 2 (if installed).	Verify proper altitude pulse train or decoded altitude on the ramp tester.			
56.	Continue to interrogate each transponder with the ramp tester through the remainder of the scale test, up to the maximum altitude.	Record the altitude reporting of each transponder at each test point in Table V and verify that the reported altitude is within MODE C tolerance of ±125 feet.			
57.	*Use the pitot-static tester to apply suction to the static system in accordance with the test points listed in Table V up to the maximum operating altitude of ▶ ___**___ feet for the aircraft. NOTE: Do not exceed the maximum altitude of the aircraft by more than 2500 feet.*				
58.	Adjust the pitot-static test set vertical speed to climb at 4000–6000 FPM between test points. Stop when the tester reaches each test point and record the indicated altitude on each altimeter.	Verify that the indicated altitude on each altimeter is within the SCALE tolerance specified in Table V.			
59.	Turn off transponder(s) after reaching maximum altitude.				
60.	Turn off transponder ramp tester after reaching maximum altitude.				
61.	End of Scale Error and Altitude Reporting Test. *Begin the hysteresis (dive) test from the maximum altitude within 15 minutes of reaching the maximum altitude.*				

** 30,000 FT - PA-46-500TP
25,000 FT - PA-46-350P

Ground Test Procedure (Sheet 15 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

Table V — Scale Error and Altitude Reporting Test (Climb at 4000–6000 FPM)

TEST POINTS Test Set Altitude	Pilot PFD Altimeter	Copilot PFD Altimeter (if installed)	Standby Altimeter (if installed)	SCALE Tolerance ±(feet)	Transponder1 Mode C Altitude (if installed)	Transponder2 Mode C Altitude (if installed)	MODE C Tolerance ±(feet)	Pass/Fail
-1,000				±20			±125	
0				±20			±125	
500				±20			±125	
1,000				±20			±125	
1,500				±25			±125	
2,000				±30			±125	
3,000				±30			±125	
4,000				±35			±125	
6,000				±40			±125	
8,000				±60			±125	
10,000				±80			±125	
12,000				±90			±125	
14,000				±100			±125	
16,000				±110			±125	
18,000				±120			±125	
20,000				±130			±125	
22,000				±140			±125	
25,000				±155			±125	
30,000				±180			±125	
35,000				±205			±125	
40,000				±230			±125	
45,000				±255			±125	
50,000				±280			±125	

Ground Test Procedure (Sheet 16 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
62.	**Hysteresis (Dive) and After Affect Test**				
63.	***Purpose.*** *This test will verify the accuracy of the altimeters after a simulated dive from maximum altitude, to 50% of maximum, to 40%, to Atmospheric Pressure, in compliance with § 43 Appendix E, Table II.* ***!!*** *Begin dive test at maximum altitude within 15 minutes of reaching maximum altitude.* *Use Table VI to record readings, calculations, and pass/fail results.*				
64.	*Descend to 50% test point.* Adjust test set to descend to the 50% test point at greater than or equal to 6000 FPM until within 3000 feet. Approach at 3000 FPM. Hold at test point for 5 minutes before taking a reading.	Verify indicated altitude is within ±75 feet of the value recorded in Table V at the same altitude after holding at the 50% test point for 5 minutes.			
65.	*Descend to 40% test point.* Adjust test set to descend to the 40% test point at greater than or equal to 6000 FPM until within 3000 feet. Approach at 3000 FPM. Hold at test point for 1 minute before taking a reading.	Verify indicated altitude is within ±75 feet of the value recorded in Table V at the same altitude after holding at the 40% test point for 5 minutes.			
66.	Descend to test point. Adjust test set to descend to Atmospheric Pressure at greater than or equal to 6000 FPM until within 3000 feet. Approach at 3000 FPM and hold.	Within 5 minutes of reaching Atmospheric Pressure, verify indicated altitude is within ±30 feet of the pitot-static test set.			

Table VI — Hysteresis Test (dive from MAX ALT at greater than or equal to 6000 FPM)

TEST POINT	Calculated Test Point Altitude (from Section 16)	Pilot PFD Altimeter	Copilot PFD Altimeter (if installed)	Standby Altimeter (if installed)	Tolerance (±feet) from the value recorded in Table V	Pass/Fail
50% test point	▶				±75 after 5 min	
40% test point	▶				±75 after 1 min	
Atmospheric Pressure	▶				±30 within 5 min	

Ground Test Procedure (Sheet 17 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
67.	**Friction Test (750 FPM Climb)**				
68.	***Purpose.*** *This test will verify the accuracy of the altimeters during a 750 FPM climb, in compliance with § 43 Appendix E, Table III.* ***Method.*** *Each static system shall be tested separately. Both the pitot and static ports of each system shall be connected to the pitot-static tester.* *All instruments in the aircraft under test shall be connected to their normal lines.* *Each altimeter in the aircraft under test will be compared to the altitude of the tester at test point altitudes, as specified in Table VII.* *Use Table VII to record all readings, calculations, and pass/fail results.*				
69.	Set pitot-static tester altimeter to 29.92 and airspeed to 0.				
70.	Press the BARO knob on PFD 1 and 2 for STD (29.92) setting.				
71.	Set the BARO on standby altimeter to 29.92.				
72.	*Use the pitot-static tester to apply suction to the static system in accordance with the test points listed in Table VII up to the maximum operating altitude of ► ________ feet for the aircraft. NOTE: Do not exceed the maximum altitude of the aircraft by more than 2500 feet.*				
73.	Adjust the pitot-static test set vertical speed to climb at 750 FPM between test points. Stop when the tester reaches each test point and record the indicated altitude on each altimeter.	Verify that the indicated altitude on each altimeter is within the FRICTION tolerance specified in Table VII.			
74.	End of Friction Test Adjust pitot-static tester to descend to atmospheric conditions.				

Ground Test Procedure (Sheet 18 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

Table VII — Friction Test (Climb at 750 FPM)

TEST POINTS Test Set Altitude	Pilot PFD Altimeter	Copilot PFD Altimeter (if installed)	Standby Altimeter (if installed)	FRICTION Tolerance ±(feet)				Pass/Fail
1,000				±70				
2,000				±70				
3,000				±70				
6,000				±70				
10,000				±80				
16,000				±90				
20,000				±100				
25,000				±120				
30,000				±140				
35,000				±160				
40,000				±180				
50,000				±250				

Ground Test Procedure (Sheet 19 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
75.	**Airspeed Test**				
76.	***Purpose.*** *This test will verify the accuracy of the airspeed indicators at pressure altitude, in compliance with § 23.1323(a).* ***Method.*** *Each pitot-static system may be tested separately or together. The pitot and static ports of each system shall be connected to the pitot-static tester.* *All instruments in the aircraft under test shall be connected to their normal lines.* *Each airspeed indicator in the aircraft under test will be compared to the airspeed Indicator of the tester at various test point speeds, as specified in Table VIII.* *Use Table VIII to record readings, calculations, and pass/fail results.*				
77.	Set pitot-static tester altimeter to 29.92.				
78.	Press the BARO knob on PFD 1 and 2 for STD (29.92) setting.				
79.	Set the BARO on standby altimeter to 29.92.				
80.	*Use the pitot-static tester to apply pressure to the pitot system in accordance with the test points listed in Table VIII up to the maximum operating speed of ► _________knots for the aircraft. NOTE: Do not exceed the maximum airspeed of the aircraft by more than 10 knots.*				
81.	Adjust the pitot-static test set to attain the speeds specified in Table VIII. Stop when the tester reaches each test point and record the indicated airspeed.	Verify that the indicated airspeed on each indicator is within the tolerance specified in Table VIII.			
82.		Verify redline displayed at the maximum airspeed per AFM at ► ____________ kts			
83.		Verify overspeed alerter (if installed) activates within ±2 knots of redline.			
84.	Decrease pitot-static tester to atmospheric conditions.				

Ground Test Procedure (Sheet 20 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

Table VIII — Indicated Airspeed (at field elevation)

Test Set Indicated Airspeed	Pilot PFD Indicated Airspeed	Copilot PFD Indicated Airspeed	Standby Airspeed Indicator	Tolerance ±KIAS	Pass	Fail
40				±5		
50				±5		
60				±5		
70				±5		
80				±5		
90				±5		
100				±5		
110				±5		
120				±5		
130				±5		
140				±5		
150				±5		
160				±5		
170				±5		
180				±6		
190				±6		
200				±6		
210				±6		
220				±7		
230				±7		
240				±7		
250				±7		
260				±8		

Ground Test Procedure (Sheet 21 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

NO	PROCEDURE	RESULTS	PILOT Pass/Fail	COPILOT Pass/Fail	STBY Pass/Fail
85.	**Shut Down**				
86.	Select EFIS off.				
87.	Shut down electrical power.				
88.	Select cabin lights off.				
89.	Disconnect Ground Power Unit from aircraft.				
90.	Disconnect pitot-static tester from aircraft.				

Note:
Remove any seals or adhesive from pitot-static ports upon completion of this test procedure.

Ground Test Procedure (Sheet 22 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

Table IX — Pressure-to-Altitude Conversion Chart

Altitude (ft)	Pressure (PSI)	Altitude (ft)	Pressure (PSI)	Altitude (ft)	Pressure (PSI)	Altitude (ft)	Pressure (PSI)
-1000	15.2348	17500	7.4915	36000	3.2966	54500	1.3549
-500	14.9634	18000	7.3389	36500	3.2183	55000	1.3227
0	14.6959	18500	7.1888	37000	3.1419	55500	1.2913
500	14.4323	19000	7.0412	37500	3.0673	56000	1.2606
1000	14.1726	19500	6.8961	38000	2.9945	56500	1.2307
1500	13.9166	20000	6.7534	38500	2.9234	57000	1.2015
2000	13.6644	20500	6.6131	39000	2.854	57500	1.173
2500	13.4159	21000	6.4752	39500	2.7862	58000	1.1451
3000	13.1711	21500	6.3397	40000	2.72	58500	1.1179
3500	12.9299	22000	6.2064	40500	2.6554	59000	1.0914
4000	12.6923	22500	6.0754	41000	2.5924	59500	1.0655
4500	12.4583	23000	5.9466	41500	2.5308	60000	1.0402
5000	12.2277	23500	5.8201	42000	2.4707		
5500	12.0007	24000	5.6958	42500	2.4121		
6000	11.777	24500	5.5736	43000	2.3548		
6500	11.5567	25000	5.4536	43500	2.2989		
7000	11.3398	25500	5.3356	44000	2.2443		
7500	11.1262	26000	5.2197	44500	2.191		
8000	10.9159	26500	5.1059	45000	2.139		
8500	10.7088	27000	4.9941	45500	2.0882		
9000	10.5049	27500	4.8843	46000	2.0386		
9500	10.3041	28000	4.7764	46500	1.9902		
10000	10.1065	28500	4.6705	47000	1.9429		
10500	9.9119	29000	4.5665	47500	1.8968		
11000	9.7204	29500	4.4644	48000	1.8518		
11500	9.5319	30000	4.3641	48500	1.8078		
12000	9.3464	30500	4.2657	49000	1.7649		
12500	9.1638	31000	4.1691	49500	1.723		
13000	8.9841	31500	4.0742	50000	1.682		
13500	8.8072	32000	3.9811	50500	1.6421		
14000	8.6332	32500	3.8898	51000	1.6031		
14500	8.462	33000	3.8001	51500	1.565		
15000	8.2935	33500	3.7121	52000	1.5279		
15500	8.1278	34000	3.6258	52500	1.4916		
16000	7.9648	34500	3.5411	53000	1.4562		
16500	7.8044	35000	3.458	53500	1.4216		
17000	7.6466	35500	3.3765	54000	1.3878		

Refer to Table IV GEOPOTENTIAL ALTITUDE, ENGLISH UNITS, contained in the document titled "U.S. Standard Atmosphere, 1976 (Stock No. 003-017-00323-0)

Ground Test Procedure (Sheet 23 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

17. Ground Test Witness

I certify that I have performed or witnessed this Ground Test Procedure on the following aircraft:

TEST LOCATION			
TYPE AIRCRAFT	SERIAL No.	REGISTRATION No.	MODE S ADDRESS (if applicable).
WITNESS NAME	SIGNATURE		DATE

Comments:

Ground Test Procedure (Sheet 24 of 24)
Meggitt MAGIC Air Data and Heading Reference System (ADAHRS) and Pitot-Static System

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