CHAPTER 27

FLIGHT CONTROLS

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* 27 - LOEP	1	Jan 17/05
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27-00-00	1	Jun 25/93
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* 27-00-00	201	Jan 17/05
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* 27-00-00	205	Jan 17/05
* 27-00-00	206	Jan 17/05
* 27-00-00	207	Jan 17/05
* 27-00-00	208	Jan 17/05
* 27-00-00	209	Jan 17/05
* 27-00-00	210	Jan 17/05
* 27-00-00	211	Jan 17/05
* 27-00-00	212	Jan 17/05
* 27-00-00	213	Jan 17/05
* 27-00-00	214	Jan 17/05
* 27-00-00	215	Jan 17/05
* 27-00-00	216	Jan 17/05
* 27-00-00	217	Jan 17/05
* 27-00-00	218	Jan 17/05
* 27-00-00	219	Jan 17/05
* 27-00-00	220	Jan 17/05
* 27-00-00	221	Jan 17/05
* 27-00-00	222	Jan 17/05
* 27-00-00	223	Jan 17/05
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* 27-00-00	225	Jan 17/05
* 27-00-00	226	Jan 17/05
* 27-00-00	227	Jan 17/05
27-10-00	1	Mar 24/95
27-10-00	2	Mar 24/95
27-10-00	3	Mar 24/95
27-10-00	4	Mar 24/95
27-10-00	5	Nov 4/83
27-10-00	6	Nov 4/83
27-10-00	7	Nov 4/83
27-10-00	8	Nov 4/83
27-10-00	9	Mar 24/95
27-10-00	10	Mar 24/95
27-10-00	11	Mar 24/95
27-10-00	12	Mar 24/95
27-10-00	13	Nov 4/83
27-10-00	14	Nov 4/83
27-10-00	15	Feb 11/00
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27-10-00	201	Feb 11/00
27-10-00	202	Feb 11/00
27-10-00	203	Feb 11/00
27-10-00	204	Feb 11/00
27-10-00	205	May 31/91
27-10-00	206	May 31/91
27-10-00	207	Jul 15/89
27-10-00	208	Jul 15/89
27-10-00	209	Jul 15/89
27-10-00	210	Jul 15/89
27-10-00	211	Feb 11/00
27-10-00	212	Feb 11/00
27-10-00	213	Feb 11/00
27-10-00	214	Feb 11/00
27-10-01	201	Jan 12/01
27-10-01	202	Jan 12/01
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27-10-01	205	Jan 12/01
27-10-02	201	Jun 22/90
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Insert latest revised pages; destroy superseded or deleted pages.

* Asterisk indicates pages revised, added, or deleted by the current revision. The portion of the text affected by the current revision is indicated by a vertical line in the outer margin of the page.

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27-10-02	205	Jun 22/90
27-10-03	201	Feb 23/90
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27-10-04	201	Mar 24/95
27-10-04	202	Mar 24/95
27-10-05	201	Jun 22/90
27-10-05	202	Jun 22/90
27-10-05	203	Jun 22/90
27-10-05	204	Jun 22/90
27-10-05	205	Jun 22/90
27-10-05	206	Jun 22/90
27-10-05	207	Jun 22/90
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27-10-06	201	Jul 15/89
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27-10-07	201	May 31/91
27-10-07	202	May 31/91
27-20-00	1	Nov 4/83
27-20-00	2	Nov 4/83
27-20-00	3	Nov 4/83
27-20-00	101	Jul 15/89
27-20-00	102	Jul 15/89
27-20-00	103	Jul 15/89
27-20-00	104	Jul 15/89
27-20-00	201	Mar 24/95
27-20-00	202	Mar 24/95
27-20-00	203	Mar 24/95
27-20-00	204	Mar 24/95
27-20-00	205	Mar 24/95
27-20-01	201	May 31/91
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27-20-06	202	Jan 10/92
27-20-07	201	Jan 12/01
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27-20-08	201	Feb 23/90
27-20-08	202	Feb 23/90
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27-30-00	205	Jun 25/93
27-30-00	206	Jun 25/93
27-30-00	207	Mar 24/95
27-30-00	208	Mar 24/95
27-30-00	209	Mar 24/95
27-30-01	201	Feb 11/00
27-30-01	202	Feb 11/00
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27-30-02	201	May 31/91
27-30-02	202	May 31/91
27-30-02	203	Oct 26/84
27-30-03	201	Jan 11/02
27-30-03	202	Jan 11/02
27-30-03	203	Jan 11/02
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27-30-03	207	Jan 11/02
27-30-04	201	Jan 10/92
27-30-04	202	Jan 10/92
27-30-04	203	Jan 10/92
27-30-04	204	Jan 10/92
27-31-00	1	Nov 4/83
27-31-00	2	Oct 26/84
27-31-00	3	Nov 4/83
27-31-00	4	Nov 4/83
27-31-00	5	Nov 4/83
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27-40-01	201	Feb 11/00
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27-40-02	202	Nov 4/83
27-40-03	201	Nov 4/83
27-40-03	202	Nov 4/83
27-40-04	201	Jun 25/93
27-40-04	202	Jun 25/93
27-40-04	203	Feb 11/00
27-40-04	204	Feb 11/00
27-40-04	205	Feb 11/00
27-40-05	201	Feb 11/00
27-40-05	202	Feb 11/00
27-40-05	203	Feb 11/00
27-40-06	201	May 31/91
27-40-06	202	May 31/91
27-41-00	1	Nov 4/83
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27-41-00	3	Nov 4/83
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27-41-00	202	Feb 11/00
27-41-00	203	Feb 11/00
27-41-00	204	Feb 11/00
27-41-00	205	Feb 11/00
27-41-01	201	Sep 25/92
27-41-01	202	Sep 25/92
27-41-01	203	Sep 25/92
27-50-00	1	May 31/91
27-50-00	2	May 31/91
27-50-00	3	May 31/91
27-50-00	4	May 31/91
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27-50-00	7	May 31/91
27-50-00	8	May 31/91
27-50-00	101	Nov 4/83
27-50-00	102	Nov 4/83
27-50-00	201	Mar 24/95
27-50-00	202	Mar 24/95
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27-50-00	205	Mar 24/95
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27-50-01	201	Mar 24/95
27-50-01	202	Mar 24/95
27-50-01	203	Feb 11/00
27-50-01	204	Feb 11/00
27-50-01	205	Feb 11/00
27-50-01	206	Feb 11/00
27-50-01	207	Feb 11/00
27-50-02	201	Sep 25/92
27-50-02	202	Sep 25/92
27-50-02	203	Sep 25/92
27-50-03	201	Nov 4/83
27-50-03	202	Nov 4/83
27-50-04	201	Nov 4/83
27-50-04	202	Nov 4/83
27-50-04	801	Nov 4/83
27-50-04	802	Nov 4/83
27-50-05	201	Jun 12/87
27-50-05	202	Jun 12/87
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27-50-05	204	Jun 12/87
27-50-05	205	Jun 12/87
27-50-05	206	Jun 12/87
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27-50-06	201	Feb 11/00
27-50-06	202	Feb 11/00
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27-51-00	8	Jun 25/93
27-51-00	9	Jun 12/87
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27-51-00	202	Mar 24/95
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27-51-01	201	Mar 24/95
27-51-01	202	Mar 24/95
27-51-01	203	Feb 11/00
27-51-01	204	Feb 11/00
27-51-01	205	Feb 11/00
27-51-01	206	Feb 11/00
27-51-02	201	Oct 26/84

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27-51-02	203	Oct 26/84
27-51-02	204	Oct 26/84
27-51-02	205	Jun 12/87
27-51-02	206	Jun 12/87
27-51-02	207	Oct 26/84
27-51-02	208	Oct 26/84
27-51-03	201	Mar 24/95
27-51-03	202	Mar 24/95
27-51-03	203	Mar 24/95
27-51-04	201	Mar 24/95
27-51-04	202	Mar 24/95
27-51-04	203	May 31/91
27-51-04	204	May 31/91
27-51-05	201	Jul 15/89
27-51-06	201	Nov 4/83
27-51-06	202	Nov 4/83
27-51-07	201	Nov 4/83
27-51-07	202	Oct 26/84
27-51-07	203	Nov 4/83
27-51-08	201	Nov 4/83
27-51-08	202	Nov 4/83
27-51-09	201	Sep 25/92
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27-51-09	203	Sep 25/92
27-51-10	201	Feb 11/00
27-51-10	202	Feb 11/00
27-60-00	1	May 31/91
27-60-00	2	May 31/91
27-60-00	3	May 31/91
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27-60-00	9	May 31/91
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27-60-00	11	May 31/91
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27-60-00	17	May 31/91
27-60-00	18	May 31/91
27-60-00	201	Feb 11/00
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27-60-00	203	Feb 11/00
27-60-00	204	Feb 11/00
27-60-00	205	Feb 11/00
27-60-00	206	Feb 11/00
27-60-01	201	Jan 11/02
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27-60-01	211	Jan 11/02
27-60-02	201	Nov 4/83
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27-60-03	201	Feb 11/00
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27-60-03	205	Feb 11/00
27-60-03	206	Feb 11/00
27-60-03	207	Feb 11/00
27-60-03	208	Feb 11/00
27-60-03	209	Nov 4/83
27-60-03	210	Nov 4/83
27-60-03	211	Nov 4/83
27-60-04	201	Mar 24/95
27-60-04	202	Mar 24/95
27-60-04	203	Mar 24/95
27-60-05	201	Nov 4/83
27-60-05	202	Nov 4/83
27-60-05	203	Nov 4/83
27-60-05	204	Nov 4/83
27-60-06	201	Sep 25/92
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27-60-08	201	May 31/91

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27-1	Inactive	Mar 24/75	27-30-00 Page 501	Mar 24/75	LJ	May 16/75 TR 27-10	LJ
27-2	Inactive	Mar 24/75	27-60-03 Page 207	Mar 24/75	LJ	Jun 30/75 Rev #3	LJ
27-3	Inactive	Apr 8/75	27-20-00 Page 2	Apr 8/75	LJ	Jun 30/75 Rev #3	LJ
27-4	Inactive	Apr 8/75	27-60-00 Page 502	Apr 8/75	LJ	Jun 30/75 Rev #3	LJ
27-5	Inactive	Apr 8/75	27-60-03 Page 205	Apr 8/75	LJ	Jun 30/75 Rev #3	LJ
27-6	Inactive	Apr 8/75	27-60-00 Page 202	Apr 8/75	LJ	Jun 30/75 Rev #3	LJ
27-7	Inactive	Apr 8/75	27-00-00 Page 201	Apr 8/75	LJ	Jun 30/75 Rev #3	LJ
27-8	Inactive	Apr 8/75	27-10-00 Page 502	Apr 8/75	LJ	Jun 30/75 Rev #3	LJ
27-9	Inactive	Apr 8/75	27-50-00 Page 3	Apr 8/75	LJ	Jun 30/75 Rev #3	LJ
27-10	Inactive	May 16/75	27-30-00 Page 501	May 16/75	LJ	Jun 30/75 Rev #3	LJ
27-11	Inactive	May 16/75	27-30-00 Page 501	May 16/75	LJ	Jun 30/75 Rev #3	LJ
27-12	Inactive	May 16/75	27-50-00 Page 204	May 16/75	LJ	Jun 30/75 Rev #3	LJ
27-13	Inactive	May 16/75	27-50-00 Page 207	May 16/75	LJ	Jun 30/75 Rev #3	LJ

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27-14	Inactive	May 30/75	27-30-00 Page 201	May 30/75	LJ	Jun 30/75 Rev #3	LJ
27-15	Inactive	Jul 15/75	27-40-00 Page 202	Jul 15/75	LJ	Oct 1/75 Rev #4	LJ
27-16	Inactive	Oct 1/75	27-40-00 Page 203	Oct 1/75	LJ	Oct 1/75 Rev #4	LJ
27-17	Inactive	Oct 1/75	27-40-00 Page 205	Oct 1/75	LJ	Oct 1/75 Rev #4	LJ
27-18	Inactive	Oct 1/75	27-40-00 Page 502	Oct 1/75	LJ	Oct 1/75 Rev #4	LJ
27-19	Inactive	Oct 1/75	27-30-00 Page 2	Oct 1/75	LJ	Oct 1/75 Rev #4	LJ
27-20	Inactive	Oct 1/75	27-30-00 Page 201	Oct 1/75	LJ	Oct 1/75 Rev #4	LJ
27-21	Inactive	Nov 2/75	27-20-00 Page 502	Nov 2/75	LJ	Dec 12/75 Rev #5	LJ
27-22	Inactive	Nov 3/75	27-20-02 Page 202	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ
27-23	Inactive	Nov 3/75	27-20-04 Page 203	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ
27-24	Inactive	Nov 3/75	27-30-00 Page 205	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ
27-25	Inactive	Nov 3/75	27-30-00 Page 207	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ
27-26	Inactive	Nov 3/75	27-30-00 Page 502	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ

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27-27	Inactive	Nov 3/75	27-30-00 Page 503	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ
27-28	Inactive	Nov 3/75	27-10-00 Page 502	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ
27-29	Inactive	Nov 3/75	27-10-00 Page 502A	Nov 3/75	LJ	Dec 12/75 Rev #5	LJ
27-30	Inactive	Feb 9/76	27-30-00 Page 503	Feb 9/76	LJ	Apr 30/76 Rev #6	LJ
27-31	Inactive	Feb 9/76	27-30-00 Page 504	Feb 9/76	LJ	Apr 30/76 Rev #6	LJ
27-32	Inactive	Jul 12/76	27-31-00 Page 4	Jul 12/76	LJ	Apr 20/79 Rev #13	LJ
27-33	Inactive	Jul 12/76	27-60-00 Page 10A	Jul 12/76	LJ	Feb 20/78 TR 27-41	LJ
27-34	Inactive	Jul 12/76	27-60-00 Page 10A	Jul 12/76	LJ	Feb 20/78 TR 27-41	LJ
27-35	Inactive	Aug 11/76	27-30-00 Page 503	Aug 11/76	LJ	Apr 20/79 Rev #13	LJ
27-36	Inactive	Aug 24/76	27-31-00 Page 210	Aug 24/76	LJ	Sep 22/76 TR 27-37	LJ
27-36	Inactive	Aug 30/76	27-31-04 Page 201	Aug 30/76	LJ	Apr 20/79 Rev #13	LJ
27-37	Inactive	Aug 30/76	27-31-00 Page 209	Aug 30/76	LJ	Apr 20/79 Rev #13	LJ
27-38	Inactive	Aug 8/77	27-31-00 Page 213	Aug 8/77	LJ	Apr 20/79 Rev #13	LJ

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27-40	Inactive	Jan 5/78	27-31-07 Page 201	Jan 5/78	LJ	Apr 20/79 Rev #13	LJ
27-41	Inactive	Feb 20/78	27-60-00 Page 1	Feb 20/78	LJ	Apr 20/79 Rev #13	LJ
27-42	Inactive	May 3/78	27-60-00 Page 4	May 3/78	LJ	Apr 20/79 Rev #13	LJ
27-43	Inactive	Jul 25/80	27-31-05 Page 203	Jul 25/80	LJ	Aug 18/80 TR 27-44	LJ
27-44	Inactive	Aug 18/80	27-31-05 Page 203	Aug 18/80	LJ	Apr 30/82 Rev #27	LJ
27-45	Inactive	Sep 26/80	27-60-00 Page 502	Sep 26/80	LJ	Apr 30/82 Rev #27	LJ
27-46	Inactive	Sep 26/80	27-60-00 Page 504	Sep 26/80	LJ	Apr 30/82 Rev #27	LJ
27-47	Inactive	Sep 26/80	27-60-00 Page 601	Sep 26/80	LJ	Apr 30/82 Rev #27	LJ
27-48	Inactive	Nov 14/82	27-50-00 Page 501	Nov 14/82	LJ	Apr 30/82 Rev #27	LJ
27-49	Inactive	Dec 22/80	27-31-04 Page 201	Dec 22/80	LJ	Apr 30/82 Rev #27	LJ
27-50	Inactive	Mar 21/81	27-31-07 Page 201	Mar 21/81	LJ	Apr 30/82 Rev #27	LJ
27-51	Inactive	Mar 27/81	27-31-07 Page 201	Mar 27/81	LJ	Apr 30/82 Rev #27	LJ

Record of Temporary Revisions

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27-52	Inactive	Aug 21/81	27-40-00 Page 501	Aug 21/81	LJ	Apr 30/82 Rev #27	LJ
27-53	Inactive	Aug 24/81	27-00-00 Page 1	Aug 24/81	LJ	Apr 30/82 Rev #27	LJ
27-54	Inactive	Feb 14/91	27-10-07 Page 201	Feb 14/91	LJ	May 31/91 Rev #53	LJ
27-55	Inactive	Mar 10/95	27-00-00 Page 210	Mar 10/95	LJ	Mar 24/95 Rev #62	LJ
27-56	Inactive	Jun 15/97	27-40-00 Page 205	Jun 15/97	LJ	Jul 11/97 TR 27-58	LJ
27-57	Inactive	Jun 15/97	27-30-01 Pg 201	Jun 15/97	LJ	Feb 11/00 Rev #70	LJ
27-58	Inactive	Jul 11/97	27-40-00 Pg 205	Jul 11/97	LJ	Feb 11/00 Rev #70	LJ
27-59	Inactive	Jul 11/97	27-40-05 Pg 201	Jul 11/97	LJ	Feb 11/00 Rev #70	LJ
27-60	Inactive	Feb 16/01	27-60-01 Page 205	Feb 16/01	LJ	Jan 11/02 Rev #71	LJ
27-61	Inactive	May 17/02	27-00-00 Page 211	May 17/02	LJ	Jan 17/05 Rev #73	LJ

FLIGHT CONTROLS - DESCRIPTION AND OPERATION

1. Description

A. The flight controls consist of control surfaces, warning systems and related mechanical and electrical systems that control the aircraft during flight.

NOTE:

- Due to the possible effects on stall characteristic, whenever maintenance as noted in items (1) thru (4) below is performed, the aircraft must be flight tested to verify aerodynamic acceptability and stall speeds. This flight test must be conducted by a pilot approved by Learjet Inc. for stall test flights. The stall warning system must be checked in accordance with the flight or ground calibration procedures in 27-31-00 of this maintenance manual when maintenance as noted in items (1) thru (5) is performed.

  (1) When maintenance requiring removal and installation, repair, or installation of a new wing leading edge is performed. This includes loosening and retightening any of the leading edge screws.
  (2) When a new tip tank strake is installed. *Applies to 35-067 and Subsequent and 36-018 and Subsequent and prior aircraft equipped with Reduced Approach Speed and/or equipped with Stall Fence (AAK 79-10 or AMK 83-5).*
  (3) When a new tip tank is installed. *Applies to 35-067 and Subsequent and 36-018 and Subsequent and prior aircraft equipped with Reduced Approach Speed and/or equipped with Stall Fence (AAK 79-10 or AMK 83-5).*
  (4) When maintenance requiring repair to, or installation of, new stall strips. *Applies to 35-279 and Subsequent and 36-045 and Subsequent and prior aircraft equipped with Stall Fence (AAK 79-10 or AMK 83-5).*
  (5) When a stall warning transducer, computer, or indicator is repaired, replaced, or calibrated.

- On *Aircraft 35-067 and Subsequent and 36-018 and Subsequent and prior aircraft equipped with Reduced Approach Speed*, replacement of the stall transducer vane is permitted in the event of a failed heater. Physical damage to the vane requires total transducer replacement because of possible damage to the transducer shaft, bearing, and potentiometer. Vane replacement requires functional test to verify that shaker and pusher actuation occurs within the prescribed tolerance of the vane plate marks.

- If angle-of-attack indicator replacement does not require an IND or Rx potentiometer adjustment, the aircraft does not require flight test/adjustment or ground calibration adjustment.

B. Control Surfaces

(1) The control surfaces and their locations are as follows:

Control Surface	Location
Ailerons	Outboard trailing edge of each wing.
Elevators	Trailing edge of horizontal stabilizer.
Rudder	Trailing edge of vertical stabilizer.
Horizontal Stabilizer	Top of and perpendicular to vertical stabilizer.
Flaps	Inboard trailing edge of each wing.
Spoilers	Upper wing surface forward of flaps.

(2) Ailerons provide lateral control of the aircraft. Aileron control is provided by dual control wheels. An aileron trim tab is installed on the left aileron to further increase lateral stability. Balance tabs are installed on both ailerons to decrease force required at the control wheel. During landing, aileron control is augmented by the spoiler system. On *Aircraft 35-279 and Subsequent, 36-045 and Subsequent, and prior aircraft modified per AMK 79-11 "Installation of Aileron Scupper Drains"*, the RH aileron incorporates two scuppers and the LH aileron incorporates three scuppers. All scuppers are mounted on the aileron lower side and provide a moisture drain for the ailerons.

(3) Elevators provide longitudinal control of the aircraft. Elevator control is by fore and aft movement of the control columns.

(4) The rudder provides directional control of the aircraft. Rudder control is manual through the rudder pedals. Directional trim is provided by a trim tab located on the rudder lower trailing edge.

(5) The horizontal stabilizer provides pitch trim control. Horizontal stabilizer control is through an electrical actuator.

(6) Flaps provide increased wing lift when partially extended during takeoff or landing approach. The increased lift provided by the extended flaps allows a lower airspeed when landing. Flaps are electrically controlled by a flap switch on the center pedestal and hydraulically operated.

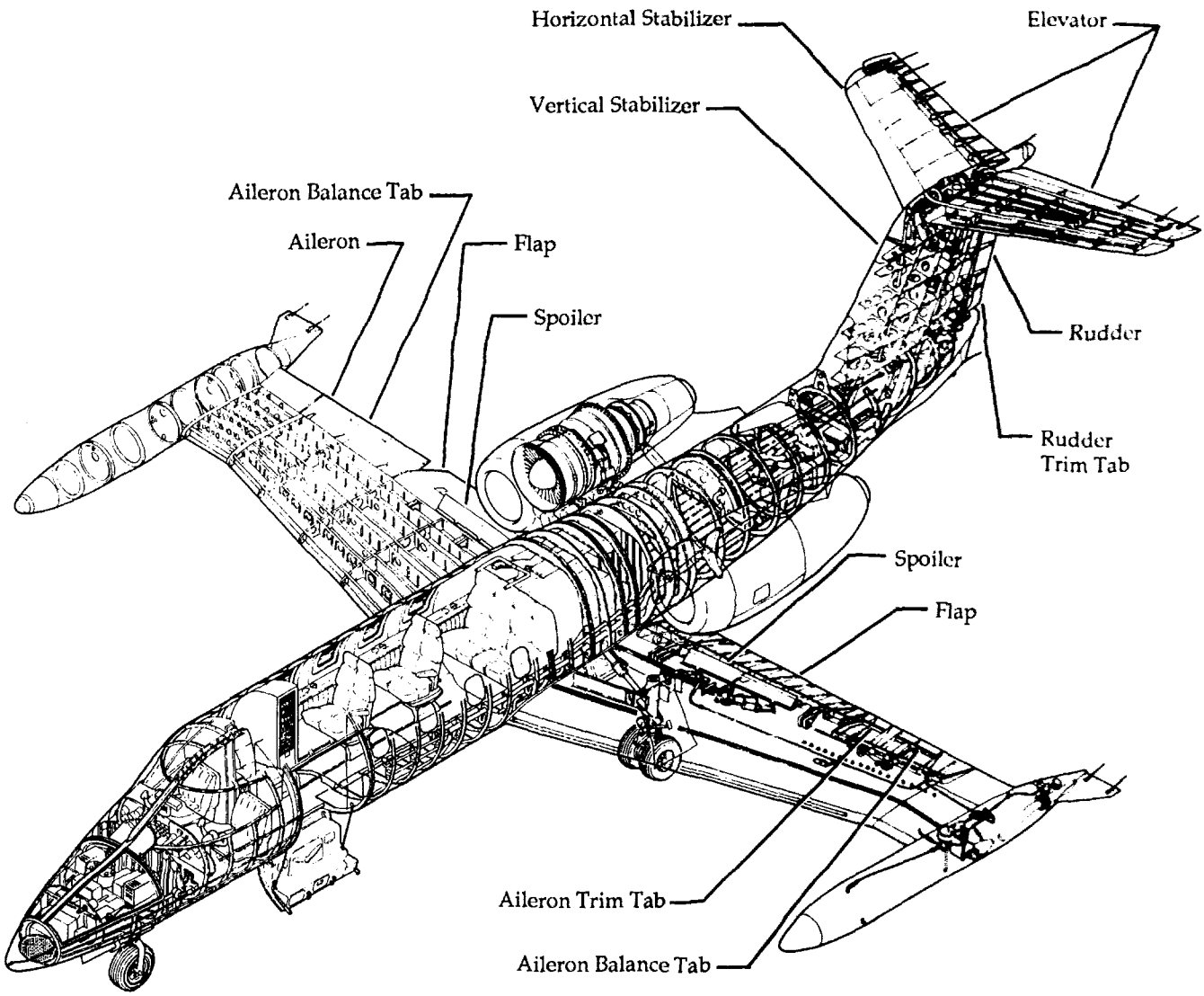
(7) Spoilers provide fast, precise speed control. Spoilers are electrically controlled by a switch on the center pedestal and hydraulically operated. During landing, the spoilers are used to augment aileron control.

C. Trim Control Surfaces

(1) The aileron trim tab provides roll trim for the aircraft. Aileron trim is obtained through the Trim and Trim Arming Switch on the control wheel. An aileron trim indicator is located on the cockpit center pedestal.

(2) The rudder trim tab provides yaw trim for the aircraft. Rudder trim is obtained through the yaw trim switch on the cockpit center pedestal.

(3) The movable horizontal stabilizer serves as a pitch trim controller. Pitch trim is obtained through the pitch trim switch on the control wheel. Secondary pitch trim is controlled through a switch on the cockpit center pedestal. A stabilizer trim indicator is located on the cockpit center pedestal. The horizontal stabilizer is also used in the autopilot and mach trim systems. (Refer to 22-00-00.)



Flight Control Surfaces
Figure 1

FLIGHT CONTROLS - MAINTENANCE PRACTICES

1. Adjustment/Test

A. Cable Tension Temperature-Load Correction (See Figure 201.)

(1) Temperature-load correction is required when checking the cable tension and rigging cables.

(2) All flight control rigging procedures should be done in a hangar, when possible, because of the cable/temperature variation. Figure 201 provides information for adjusting the cable tension loads for temperature variations.

(3) Temperatures shown in Figure 201 are for ambient air, not airframe components.

(4) If the aircraft has been flown at altitude, allow 40 to 60 minutes for stabilization of the airframe temperatures before rigging the cable tensions.

(5) Rigging may be accomplished with the aircraft on jacks without affecting cable tensions.

(6) Any approved loading configuration will not affect the cable tension rigging.

B. Tensiometer Adapter Calibration (See Figure 202.)

NOTE: The tensiometer adapter is intended for use on servo cables in areas that are inaccessible for normal tensiometer usage.

(1) Get the necessary tools and equipment.

NOTE: You can use equivalent alternatives for these items:

NAME	PART NUMBER	MANUFACTURER	USE
Dial Type Torque Wrench (0 to 50 in lb.) [0 to 5.64 Nm]		Commercially Available	Measure the cable tensions.
Tensiometer Adapter	2471007-7	Learjet Inc. Wichita, KS	Measure cable tensions.
Tension Adapter Calibration Cable	3170029-1 thru 3170029-4	Learjet Inc. Wichita, KS	Calibrate tensiometer and torque wrench.
Angles (Two required)	3170029-22	Learjet Inc. Wichita, KS	Tensiometer calibration.
Weights	3170029-23 thru 3170029-28	Learjet Inc. Wichita, KS	Tensiometer calibration.

(2) Calibration method is as follows:

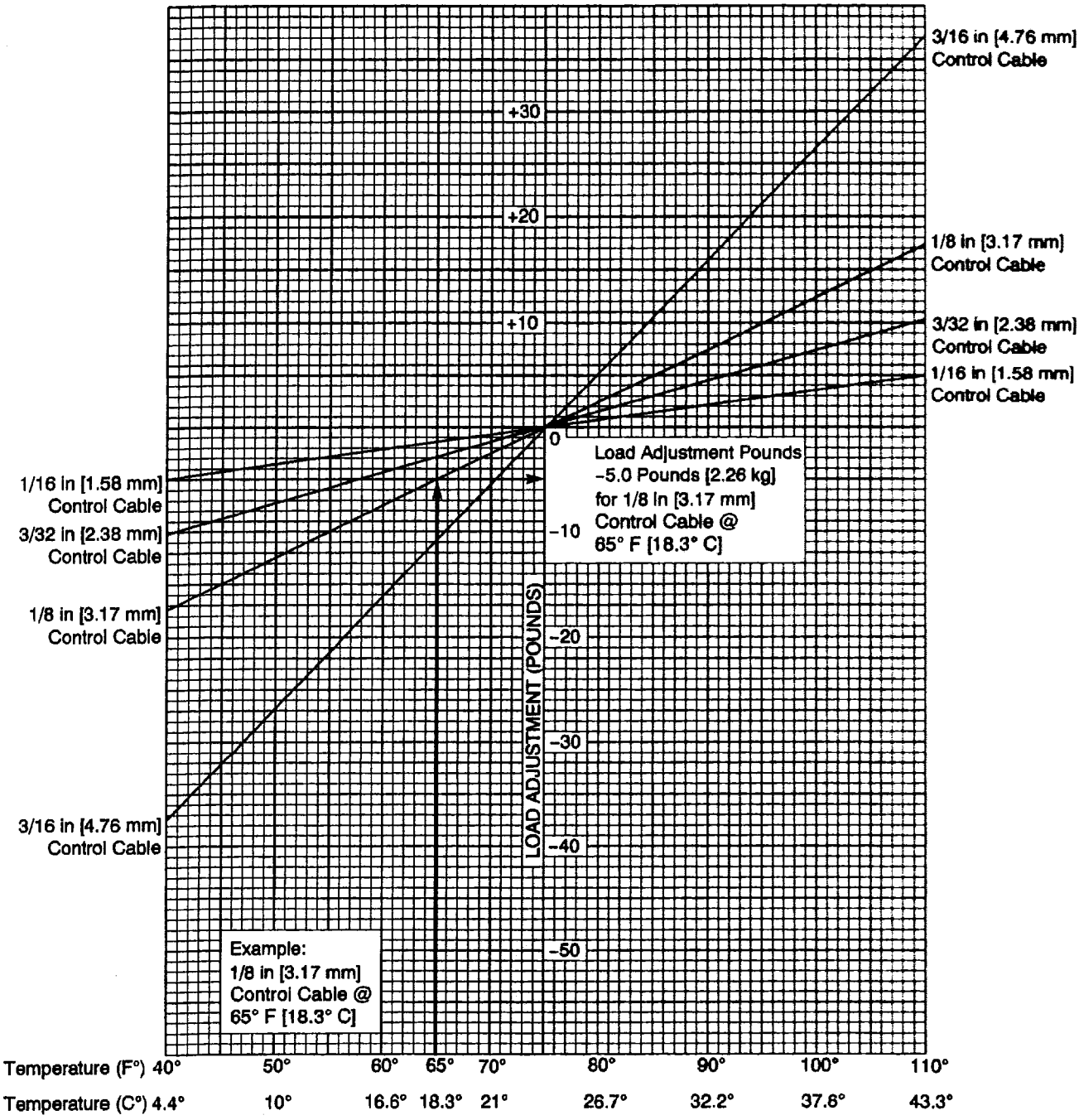
NOTE: Blanks are provided in Figure 202 to enter the different torque wrench values corresponding to both the cable tension and the temperature.

(a) Set up the following tensions: 15, 20, 25, 30, 35, and 40 pounds [6.8, 9.1, 11.3, 13.6, 15.9, and 18.1 kg] on 1/16 inch [1.58 mm], 3/32 inch [2.38 mm], 1/8 inch [3.17 mm], and 3/16 inch [4.76 mm] cables.
(b) Install the tensiometer adapter and torque wrench on the cable.
(c) Turn the torque wrench until the third arm of the tensiometer adapter contacts the cable.

NOTE: It is extremely important that maintenance personnel note the contacting pressure of the adapter third arm with the cable. Any overpressure or under pressure will result in cable tension error.

(d) Repeat the procedure a minimum of three to five times to make sure of the correct torque value.
(e) Record the torque wrench value in tensiometer adapter calibration chart.
(f) Repeat steps (b) and (c) on the remaining cable sizes, under the required tensions.
(g) Enter the torque wrench values in the appropriate blanks.

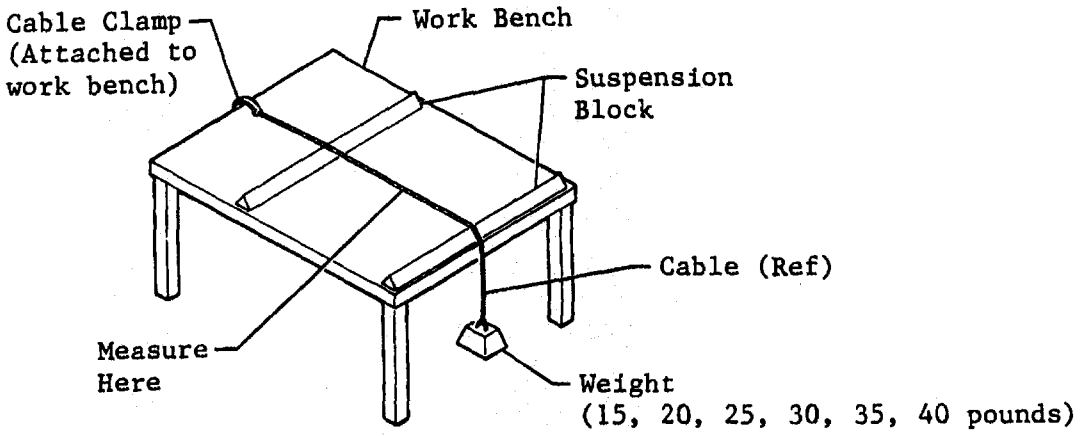
(3) Conversion method is as follows:
(a) Place the tensiometer adapter on the torque wrench. An extension may be used if necessary.
(b) Install the tensiometer adapter on the cable. (See Figure 202.)
(c) Record the torque wrench reading at the point where the third arm (one without groove) just contacts the cable.
(d) Convert the torque wrench reading to cable tension by using the values previously determined and noted in the tensiometer adapter calibration chart.



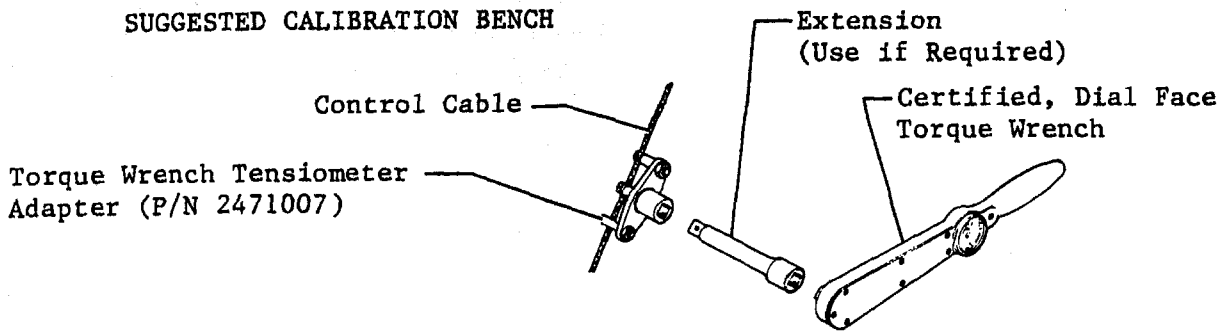
Cable Tension Temperature-Load Correction
Figure 201

CABLE TENSION		15 lbs (6.8 kg)	20 lbs (9.1 kg)	25 lbs (11.3 kg)	30 lbs (13.6 kg)	35 lbs (15.9 kg)	40 lbs (18.1 kg)
1/16-inch (1.5875 mm) Cable	0° F (-17° C) 70° F (21° C) 100° F (37° C)						
3/32-inch (2.3813 mm) Cable	0° F (-17° C) 70° F (21° C) 100° F (37° C)						
1/8-inch (3.175 mm) Cable	0°F (-17° C) 70° F (21° C) 100° F (37° C)						
3/16-inch (4.7625 mm) Cable	0° F (-17° C) 70° F (21° C) 100° F (37° C)						

TENSIOMETER ADAPTER CALIBRATION CHART



SUGGESTED CALIBRATION BENCH



Tensiometer Adapter and Torque Wrench Calibration
Figure 202 (Sheet 1 of 2)

CONTROL CABLE SIZE		
System	**Size**	**Construction**
Autopilot Roll Actuator	1/16-inch [1.6 mm]	7x7
Autopilot Pitch Actuator	3/32-inch [2.4 mm]	7x19
Yaw Damper Actuator (Pri)	3/32-inch [2.4 mm]	7x7
Yaw Damper Actuator (Sec)	3/32-inch [2.4 mm]	7x7
Aileron Control System		
Control Wheel Interconnect	1/8-inch [3.2 mm]	7x19
Control Wheel to Frame 15	1/8-inch [3.2 mm]	7x19
Frame 15 to Aft Sector	1/8-inch [3.2 mm]	7x19
Aft Sector to Aileron Sector	3/16-inch [4.8 mm]	7x19
Aileron-Rudder Interconnect	1/16-inch [1.6 mm]	7x7
Rudder Control System	3/16-inch [4.8 mm]	7x19
Elevator Control System	1/8-inch [3.2 mm]	7x19
Flap Control System	3/16-inch [4.8 mm]	7x19

Tensiometer Adapter and Torque Wrench Calibration
Figure 202 (Sheet 2 of 2)

C. Aileron Trim Tab Balancing (See Figure 203.)

NOTE: The aileron trim tab, when static-balance-checked, must have the pushrod bracket installed and painted but without the pushrod attaching parts attached.

(1) Get the necessary tools and equipment.

NOTE: *You can use equivalent alternatives for these items:*

NAME	PART NUMBER	MANUFACTURER	USE
Balance Fixture	3170035-1	Learjet Inc. Wichita, KS	Balance the aileron trim tab.
Known Weight 1 to 1.5 lbs. [0.45 to 0.69 kg]		Commercially Available	Balance aileron trim tab.

(2) Disconnect electrical power from the aircraft.
(3) *Remove the aileron trim tab from the aircraft. (Refer to 27-10-02.)*
(4) Make sure that the bench and rise are level. Clamp the balance fixture in the vise so that the knife edges are level.
(5) If necessary, remove the trim tab hinge pin and loose hinge. Reinstall the hinge pin in the trim tab hinge.
(6) Place the trim tab on the balance fixture with the pushrod bracket facing down. Make sure that the trim tab is free to turn on its hinge pin and balance the fixtures knife edges.
(7) Static-balance-check the trim tab as follows:

NOTE: *On Aircraft 35-002 thru 35-514 and 36-002 thru 36-053 not modified per SSK938, or AMK83-3*, the aileron trim tab balance limits are 0.0 to 0.1 in-lb [0.00 to 0.01 Nm] over-balance (nose heavy).

On Aircraft 35-515 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per SSK938, the aileron trim tab balance limits are 1.15 to 1.30 in-lb [0.13 to 0.15 Nm] overbalance (nose heavy).

On Aircraft 35-002 thru 35-514 and 36-002 thru 36-053 modified per AMK83-3, the aileron trim tab balance limits are 0.85 to 1.0 in-lb [0.10 to 0.11 Nm] overbalance (nose heavy).

(a) Move a known weight along the upper surface of the trim tab until the trim tab is balanced to a level condition (perpendicular to hinge centerline).
(b) Measure the distance from the centerline of the hinge to the center of gravity of known weight.
(c) Multiply this distance, by the quantity of known weight, to obtain the overbalance.

(d) If the overbalance is not within prescribed limits, correct the condition by adding or subtracting material from the trim tab balance weight. Then, static-balance-check the trim tab again to make sure that the proper overbalance is obtained.

(8) Record the overbalance and date on the inside rib surface of the trim tab.

(9) Remove the trim tab from the balance fixture.

(10) Touch up the paint on the trim tab if necessary. (Refer to 20-50-00.)

(11) Install the aileron trim tab on the aircraft. (Refer to 27-10-04.)

(12) Connect electrical power to the aircraft. (Refer to 24-40-00.)

(13) Test the Aileron Trim Tab System for Proper Operation. (Refer 27-10-00.)

D. Aileron Balancing (See Figure 204.)

NOTE: The aileron control surfaces are designed with an overbalance (nose heavy) condition. This amount of overbalance and unit weight is recorded on the inboard rib surface of the aileron.

Whenever a new flight control surface is to be installed; or after a major repair, or after repainting of an area or areas totaling over 12 sq in [74.41 sq cm] has been accomplished on the control surface; it must be static-balance-checked.

Control surface balancing must be accomplished as close to a draft-free environment as possible.

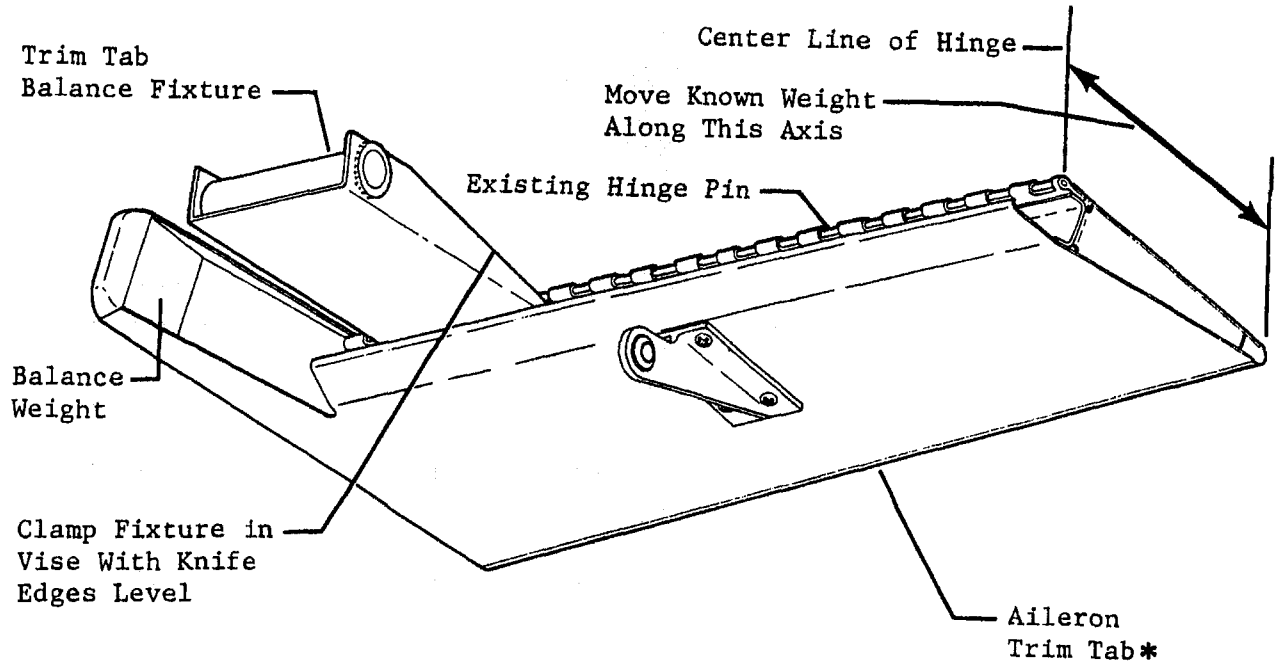
The aileron, when static-balance-checked, must have the trim tab, balance tab and push-rod installed, and be final painted.

Balancing procedures for both the LH and RH aileron are identical except as noted for the LH aileron and trim tab.

(1) Get the necessary tools and equipment.

NOTE: You can use equivalent alternatives for these items:

NAME	PART NUMBER	MANUFACTURER	USE
Balance Fixture (Two required.)	3170036-1	Learjet Inc. Wichita, KS	Balance the aileron.
Known Weight 1 to 1.5 lbs. [0.45 to 0.69 kg]		Commercially Available	Balance the aileron.
Screws	AN509-10R15	Learjet Inc. Wichita, KS	Attach lead weights to the aileron.
Lead Weights	2424002-1 and -3	Learjet Inc. Wichita, KS	Balance the aileron.



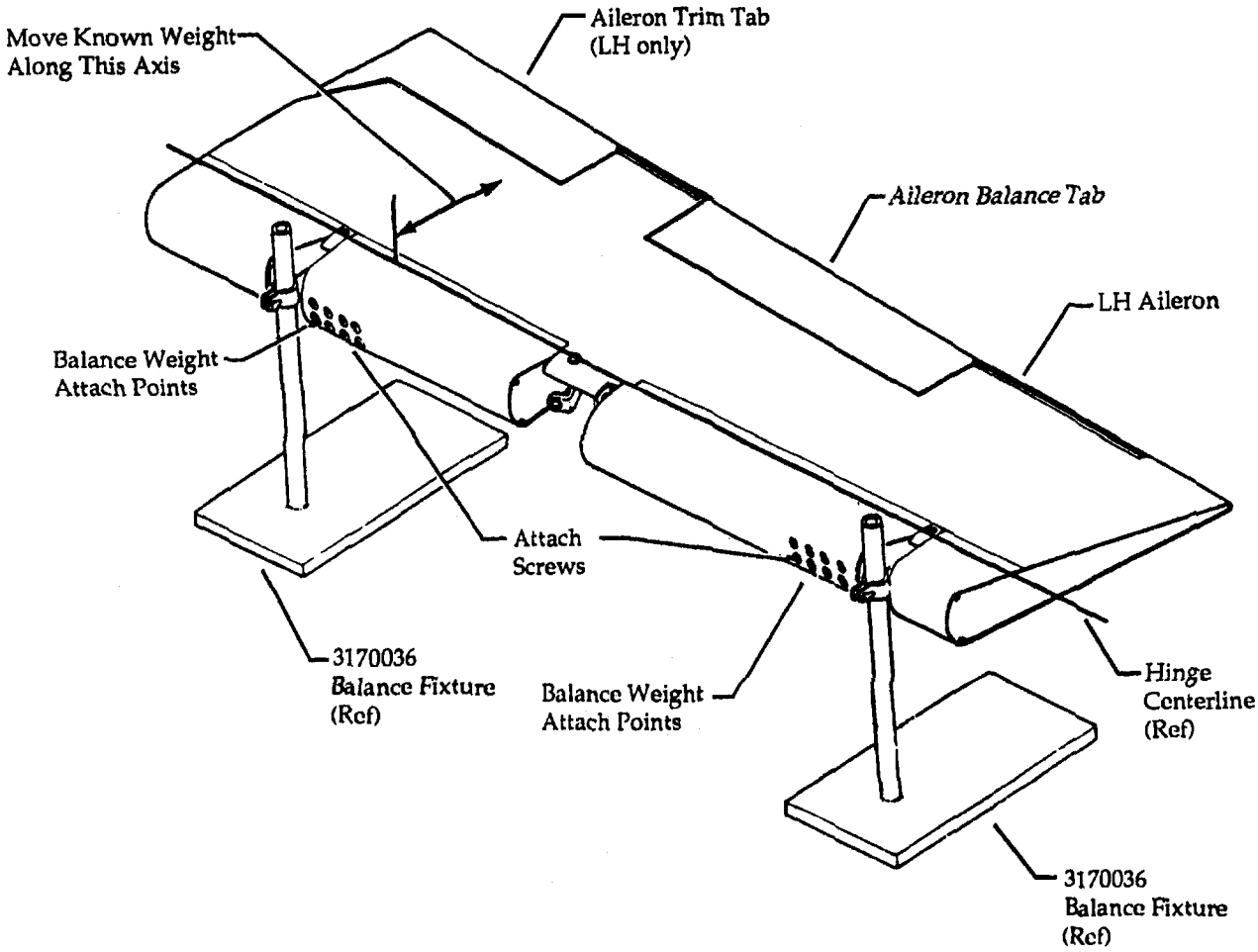
CONTROL SURFACE	BALANCE LIMITS	EFFECTIVITY
Aileron Trim Tab	0.0 to 0.1 Inch-Pounds (Overbalance) (0.00 to 0.01 Nm [Overbalance])	*On Aircraft 35-002 thru 35-514 and 36-002 thru 36-053 not modified per SSK938, "Replacement of Aileron Assembly" or AMK83-3, "Replacement of Aileron Trim Tab Balance Weight".*
	1.15 to 1.30 Inch-Pounds (Overbalance) (0.13 to 0.15 Nm [Overbalance])	*On Aircraft 35-515 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per SSK938, "Replacement of Aileron Assembly".*
	0.85 to 1.0 Inch-Pounds (Overbalance) (0.10 to 0.11 Nm [Overbalance])	*On Aircraft 35-002 thru 35-514 and 36-002 thru 36-053 modified per AMK83-3, "Replacement of Aileron Trim Tab Balance Weight".*

Aileron Trim Tab Balancing
Figure 203

(2) Disconnect electrical power from the aircraft.
(3) Make sure that the balance fixtures are free standing and level. Knife edges must be adjusted to an equal height and level.
(4) Remove the aileron from the aircraft. (Refer to 27-10-01.)
(5) Install all the previously removed hinge bolts and access covers on the aileron.
(6) If the trim tab (LH aileron) is replaced, painted, or repaired do the steps that follow:
  (a) Remove the trim tab (LH aileron) from the aircraft. (Refer to 27-10-02.)
  (b) Static-balance-check the aileron trim tab. (Refer to 27-00-00.)
  (c) Install the trim tab on the aileron after static-balance-check. (Refer to 27-10-01.)
(7) Make sure that trim tab and aileron parts are attached.
(8) Set the trim tab to the neutral position on the aileron and tape in place.
(9) Position the balance tab to the neutral position on the aileron and tape in place.
(10) Remove the aileron yokes if desired.

NOTE: The aileron yokes do not have to be removed for aileron balancing.

(11) Set the aileron as it would be on aircraft, then position the aileron on the balance fixtures with the hinge bolts resting on the knife edges.
(12) Static-balance-check the aileron as follows:
  (a) Aileron balance limits with the yokes removed are:
    1) Unpainted - 16.5 to 17.5 in-lb [1.9 to 2.0 Nm] overbalance (nose heavy).
    2) Painted - 15 to 16 in-lb [1.7 to 1.8 Nm] overbalance (nose heavy).
  (b) Aileron balance limits with the yokes attached are:
    1) Unpainted - 17.25 to 18.25 in-lb [1.95 to 2.06 Nm] overbalance (nose heavy).
    2) Painted - 15.75 to 16.75 in-lb [1.78 to 1.89 Nm] overbalance (nose heavy).
  (c) If the yoke is attached while balancing, the yoke must be in the full forward position.
  (d) Move a known weight, aft from the hinge centerline, along the upper surface of the aileron until the aileron is balanced to a level condition (perpendicular to hinge centerline).
  (e) Measure from the hinge centerline to the center of gravity of the known weight.
  (f) Multiply this distance, by the quantity of known weight, to obtain the overbalance.
  (g) If the overbalance is not within the prescribed limits, correct the condition by adding or subtracting balance weights from the inside aileron's leading edge. Static-balance-check the aileron again to make sure that the proper overbalance is obtained.
(13) Remove the aileron from the balance fixtures.
(14) Record the overbalance and date on the inboard rib surface of the aileron.
(15) Touch up the paint on the aileron and install on the aircraft. (Refer to 20-50-00.)
(16) Connect the electrical power to the aircraft.
(17) Install the aileron on the aircraft. (Refer to 27-10-01.)



CONTROL SURFACE	BALANCE LIMITS
Aileron with Yoke Removed	Unpainted: 16.5 to 17.5 Inch-Pounds (1.9 to 2.0 Nm) over balance. (Nose heavy)
	Painted: 15 to 16 Inch-Pounds (1.7 to 1.8 Nm) overbalance. (Nose heavy)

CONTROL SURFACE	BALANCE LIMITS
Aileron *with* Yoke Attached In Full Forward Postion	Unpainted: 17.25 to 18.25 Inch-Pounds (1.95 to 2.06 Nm) over-balance. (Nose heavy)
	Painted: 15.75 to 16.75 Inch-Pounds (1.78 to 1.89 Nm) over-balance. (Nose heavy)

LH AILERON (SHOWN)

M60-270000-204-01

Aileron Balancing
Figure 204

E. Rudder Balancing (See Figure 205.)

NOTE: The rudder control surface is designed with an overbalance (nose heavy) condition. This amount of overbalance and unit weight is recorded on the bottom rib surface of the rudder.

Whenever a new flight control surface is to be installed; or after a major repair, or after repainting of an area or areas totaling over 12 square inches [74.41 sq cm] has been accomplished on any flight control surface; it is essential that the control surface effected be static-balance-checked.

Control surface balancing must be accomplished in as near a draft-free environment as possible.

All control surfaces will be balanced as if they were installed on aircraft. The rudder, when static-balance-checked, must have the trim tab sector and bearing block installed and be final painted.

(1) Get the necessary tools and equipment.

NOTE: You can use equivalent alternatives for these items:

NAME	PART NUMBER	MANUFACTURER	USE
Balance Fixture (Two required.)	3170036-1	Learjet Inc. Wichita, KS	Balance the rudder.
Known Weight 1 to 1.5 lbs. [0.45 to 0.69 kg]		Commercially Available	Balance the rudder.
Screws	AN509-10R15	Learjet Inc. Wichita, KS	Attach lead weights to the rudder.
Lead Washers	2424035-1 and -2	Learjet Inc. Wichita, KS	Balance the rudder.
Weight Tray		Commercially Available	Balance the rudder.

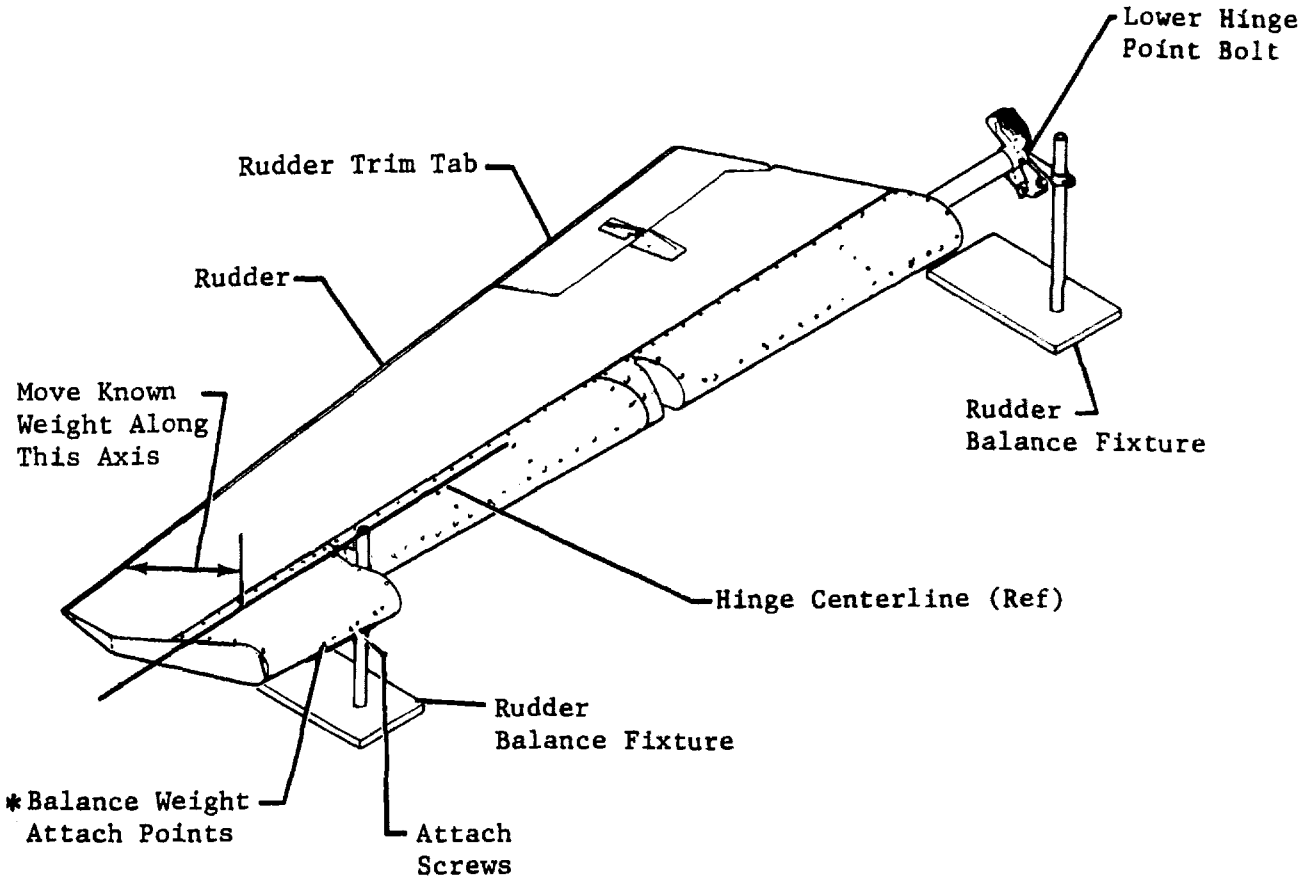
(2) Disconnect electrical power from the aircraft.
(3) Make sure that the balance fixtures are free standing and level. Knife edges must be adjusted to an equal height and level.
(4) Remove the rudder from the aircraft. (Refer to 27-20-01.)
(5) Install all the hinge bolts and access covers on the rudder.
(6) If the rudder trim tab is replaced, painted, or repaired do the steps that follow:
  (a) Remove the rudder trim tab from the rudder. (Refer to 27-20-03.)
  (b) Static-balance-check the rudder trim tab. (Refer to 27-00-00.)
  (c) Install the rudder trim tab on the rudder. (Refer to 27-20-03.)

(7) Make sure that all the parts are installed on the trim tab and rudder. Position the trim tab in neutral on the rudder and tape in place.
(8) Set the rudder on the balance fixtures so that the upper hinge bolt and bolt through the rudder torque tube rests on the knife edges.
(9) Static-balance-check the rudder as follows:

NOTE: The rudder balance limits are 2.0 to 2.5 in-lb [0.2 to 0.3 Nm] overbalance (nose heavy).

(a) Move a known weight, aft from the hinge centerline, along the upper surface of the rudder until the rudder is balanced to a level condition (perpendicular to hinge centerline).
(b) Measure from the hinge centerline to the center of gravity of the known weight.
(c) Multiply this distance, by the quantity of the known weight, to obtain the overbalance.
(d) If the overbalance is not within the prescribed limits, correct the condition by adding or subtracting the balance weights from inside the rudders leading edge. Static-balance-check the rudder again to make sure that the proper overbalance is obtained.

(10) Remove the rudder from the balance fixtures.
(11) Record the overbalance and date on the bottom rib surface of the rudder.
(12) Touch up the paint on the rudder. (Refer to 20-50-00.)
(13) Install the rudder on the aircraft. (Refer to 27-20-01.)
(14) Connect electrical power to the aircraft.
(15) Do the Rigging of the Rudder Control System. (Refer to 27-20-00.)



CONTROL SURFACE	BALANCE LIMITS
Rudder	2 to 2.5 Inch-Pounds (Overbalance) (0.2 to 0.3 Nm [Overbalance])

(TYPICAL)

Rudder Balancing
Figure 205

F. Rudder Trim Tab Balancing (See Figure 206.)

NOTE: The rudder trim tab, when static-balanced-checked, must have the pushrod bracket installed and painted but without the pushrod attaching parts installed.

(1) Get the necessary tools and equipment.

NOTE: You can use equivalent alternatives for these items:

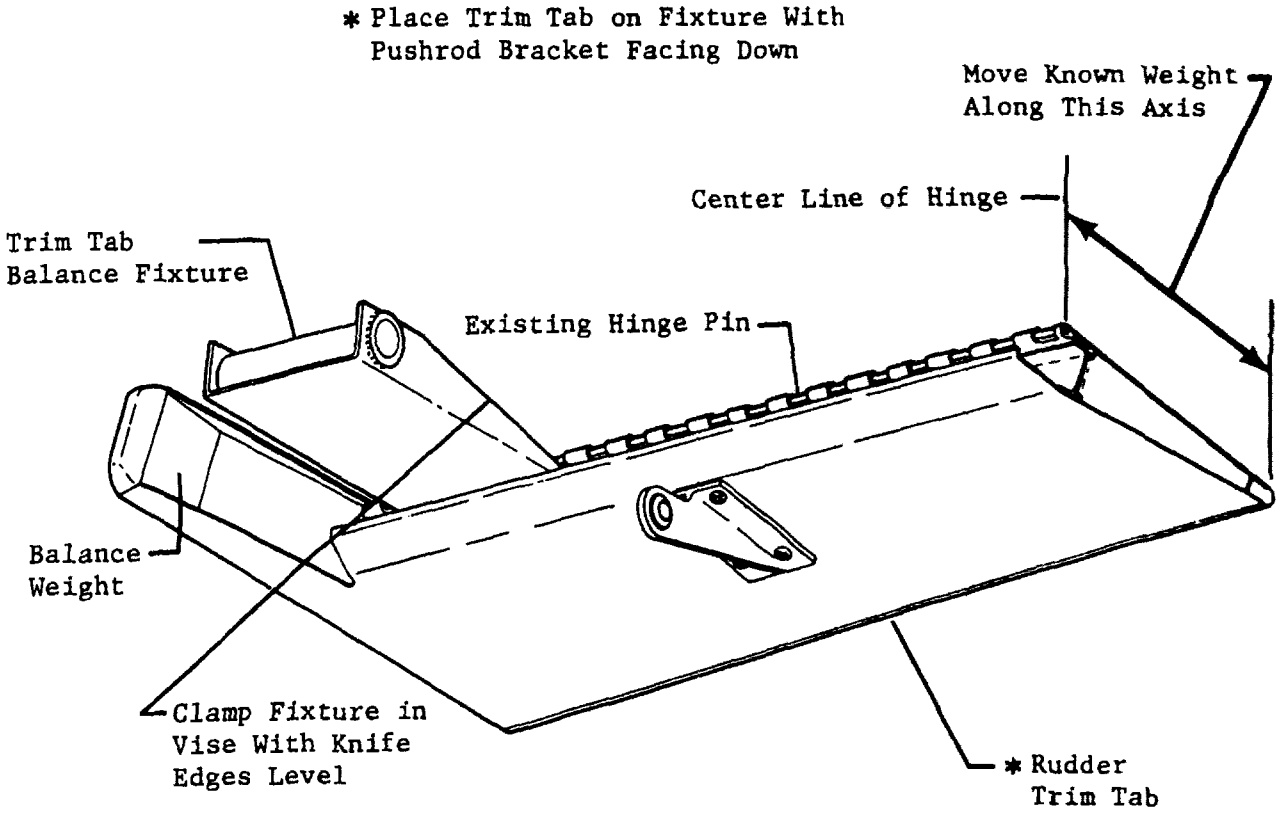
NAME	PART NUMBER	MANUFACTURER	USE
Balance Fixture	3170035-1	Learjet Inc. Wichita, KS	Balance the rudder trim tab.
Known Weight 1 to 1.5 lbs. [0.45 to 0.69 kg]		Commercially Available	Balance rudder trim tab.

(2) Remove the rudder trim tab from the aircraft. (Refer to 27-20-03.)
(3) Make sure that the bench and vise are level. Clamp the balance fixture in the vise so the knife edges are level.
(4) Remove the trim tab hinge pin and loose hinge (previously riveted to rudder control surface).
(5) Install the hinge pin in the trim tab hinge.
(6) Put the trim tab on the balance fixture with the pushrod bracket facing down. Make sure that the trim tab is free to turn on its hinge pin and balance fixtures knife edge.
(7) Static-balance-check the trim tab as follows:

NOTE: Trim tab balance limits are -0.5 to +0.5 in-lb [-0.05 to +0.05 Nm] static balance.

(a) Move a known weight along the upper surface of the trim tab until the trim tab is balanced to a level condition (perpendicular to hinge centerline).
(b) Measure the distance from the centerline of the hinge to the center of gravity of the known weight.
(c) Multiply this distance, by the known weight, to obtain the overbalance.
(d) If the overbalance is not within the prescribed limits, correct the condition by adding or subtracting material from the tab balance weight.
(e) Check the static-balance of the trim tab again to make sure that the correct overbalance is obtained.

(8) Remove the trim tab from the balance fixture.
(9) Record the overbalance and date on the inside rib surface on the trim tab.
(10) Touch up the paint on the elevator. (Refer to 20-50-00.)
(11) Install the rudder trim tab on the rudder. (Refer to 27-20-03.)



CONTROL SURFACE	BALANCE LIMITS
Rudder Trim Tab	±0.5 Inch-Pound (±0.05 Nm)

Rudder Trim Tab Balancing
Figure 206

G. Elevator Balancing (See Figure 207.)

NOTE: The elevator control surfaces are designed with an overbalance (nose heavy) condition. This amount of overbalance and unit weight is recorded on the inboard rib surface of the elevator.

Whenever a new flight control surface is to be installed; or after a major repair, or after repainting of an area or areas totaling over 12 square inches [74.41 sq cm] has been accomplished on any flight control surface; it is essential that the control surface effected be static-balanced-checked.

Control surface balancing must be accomplished in as near a draft-free environment as possible.

All control surfaces will be balanced as if they were installed on the aircraft. The elevator must have the elevator bellcrank and static discharger installed and be final painted.

Balancing procedures for both the LH and RH elevators are identical.

(1) Get the necessary tools and equipment.

NOTE: You can use equivalent alternatives for these items:

NAME	PART NUMBER	MANUFACTURER	USE
Balance Fixture (Two required.)	3170036-1	Learjet Inc. Wichita, KS	Balance the elevator.
Known Weight 1 to 1.5 lbs. [0.45 to 0.69 kg]		Commercially Available	Balance the elevator.
Screws	AN509-10R15	Learjet Inc. Wichita, KS	Attach lead weights to the elevator.
Lead Washers	2424035-1 and -2	Learjet Inc. Wichita, KS	Balance the elevator.
Weight Tray		Commercially Available	Balance the elevator.

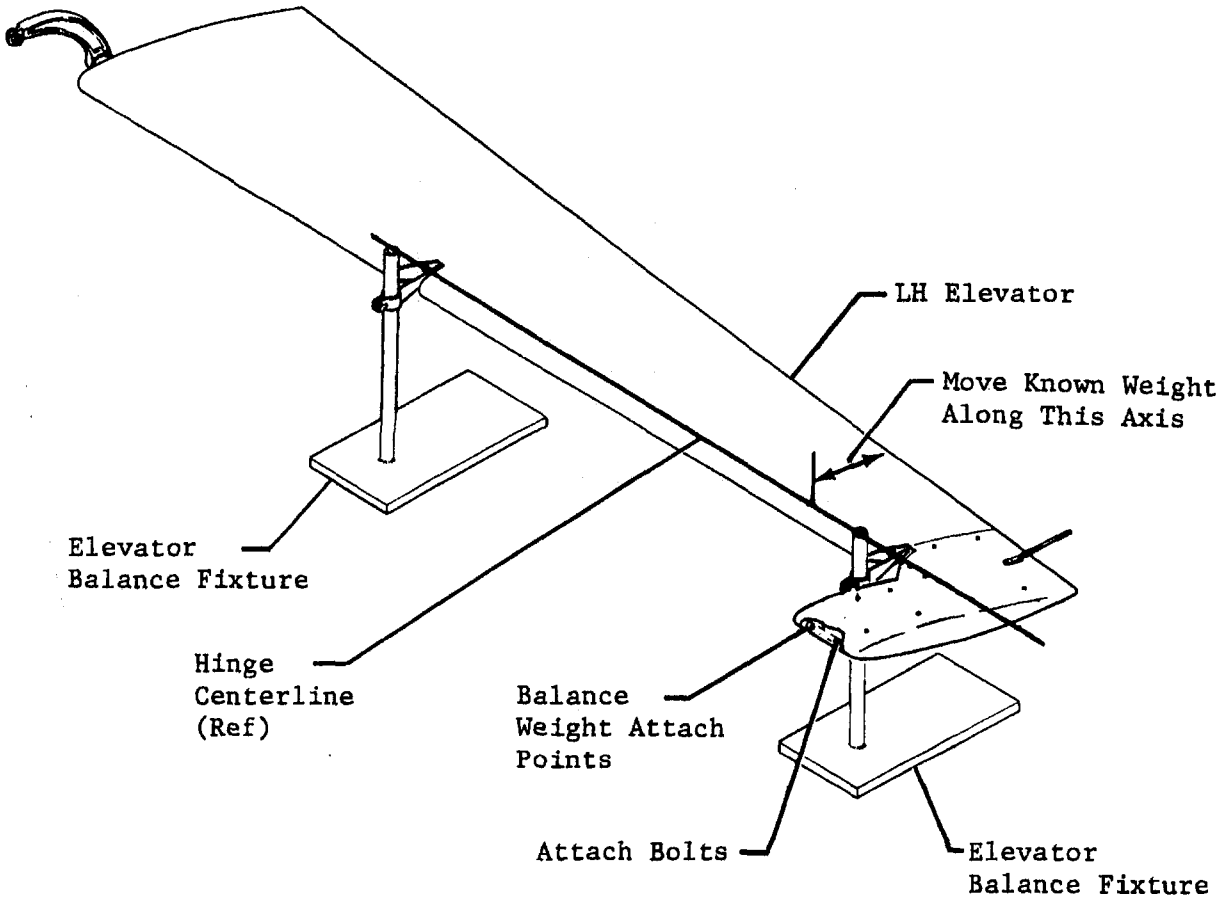
(2) Make sure that the balance fixtures are free standing and level. Knife edges must be adjusted to an equal height and level.
(3) Remove the elevator from the aircraft. (Refer to 27-30-03.)
(4) Install all hinge bolts and access covers on the elevator.
(5) Orient the elevator as it would be on the aircraft, then position the elevator on the balance fixtures with the hinge bolts resting on the knife edges.

(6) Static-balance-check as follows:

NOTE: Elevator balance limits are 10.0 to 10.5 in-lb [1.1 to 1.2 Nm] overbalance (nose heavy).

(a) Move a known weight, aft from the hinge centerline, along the upper surface of the elevator until the elevator is balanced to a level condition (perpendicular to hinge centerline).
(b) Measure from the hinge centerline to the center of gravity of known weight.
(c) Multiply this distance, by the known weight, to obtain the overbalance.
(d) If the overbalance is not within the prescribed limits, correct the condition by adding or subtracting the balance weights from inside the elevators leading edge. Static-balance-check the elevator again to make sure that the proper overbalance is obtained.

(7) Remove the elevator from the balance fixtures.
(8) Record the overbalance and date on the inboard rib surface of the elevator.
(9) Touch up the paint on the elevator. (Refer to 20-50-00.)
(10) Connect electrical power to the aircraft. (Refer to 24-40-00.)
(11) Install the elevator on the aircraft. (Refer to 27-30-03.)



CONTROL SURFACE	BALANCE LIMITS
Elevator	10.0 to 10.5 Inch-Pounds (1.1 to 1.2 Nm) (Overbalance)

LH ELEVATOR (SHOWN)

(TYPICAL)

Elevator Balancing
Figure 207

2. Inspection/Check

A. Control Cable System Inspection (See Figures 208 and 209.)

NOTE: Refer to Chapter 5 for the current inspection interval for the Control Cable System Inspection.

(1) The aircraft control cable systems are subject to a variety of environmental conditions and forms of deterioration. These conditions may be easy to recognize, such as strand or wire breakage, or the not-so-readily visible type of wear, corrosion, and/or distortion.

(2) Failure of the control cables may be progressive over periods of extended use. Some broken wires may be evident soon after placing the cables into service due to some wires being under great tension or due to variation in wire hardness. During normal service, additional broken wires will be found.

(3) Control cable strength is as much as 50% in excess of design loads so control cables may have a few broken wires and still be capable of carrying the design load of the cable.

(4) The control cables used in the aircraft are 1/16 inch [1.58 mm], 3/32 inch [2.38 mm], 1/8 inch [3.275 mm], and 3/16 inch [4.76 mm] in diameter. Cables that are of the 7 x 7 construction, contain 7 strands of 7 wires each totaling 49 wires. Cables that are of the 7 x 19 construction, contain 7 strands of 19 wires each totaling 133 wires.

(5) Critical areas for possible wire breakage are in those sections of the cable which pass through the fairleads, at control sectors, and around pulleys or capstans.

(a) Examine each cable for broken wires by passing a cloth along the length of each cable. This will detect broken wires because the cloth will snag on the broken wires. If snags are found, closely examine the cable to determine the full extent of the damage.

(b) Using a 7 power or greater magnifying glass, examine each cable for broken wires which may not protrude or stick out of the strand. Broken wires of this type will appear as a hairline crack in the wire.

(6) The absence of snags is not positive evidence that broken wires do not exist. Often broken wire is not apparent unless the cable is removed and bent into a loop, thus forcing broken wires to protrude from the cable.

(a) External Wear Patterns - Wear will normally extend along the cable equal to the distance the cable moves at that location and may occur on one (1) side of the cable only or on its entire circumference.

(b) Internal Cable Wear - As wear is taking place on the exterior surface of a cable, the same condition is taking place internally, particularly in those sections of the cable which pass over pulleys and quadrants. This condition is not easily detected unless wires of the cable become separated. This wear is a result of relative motion between the inner wire surfaces. In some instances this wear can be greater than that occurring on the outer surface of the cable.

(c) Corrosion - Any cable which has a broken wire in a section not in contact with pulleys or fairleads should be carefully inspected for corrosion. It may be necessary to remove and bend the cable to properly inspect it for internal wire corrosion. This condition is usually not evident on surface of cable.

(7) Inspect pulleys for roughness, sharp edges, and presence of foreign material embedded in pulley grooves. Examine the pulley bearings to make sure that they turn smoothly, are free of flat spots, dirt, and paint spray. Pulleys which turn through a small arc should periodically be turned to pro-

vide a new bearing surface for the cable. Maintain pulley alignment to prevent the cable from riding on pulley flanges and rubbing against the cable guards or adjacent structure.

(a) Pulley Wear Patterns - Various cable system malfunctions may be detected by analyzing pulley conditions. These include such discrepancies as too much tension, misalignment of pulley bearing, and size mismatch. Examples of these conditions are shown in Figure 208.

B. Control Cable Damage Limits (See Figure 209.)

(1) Replace any 7 x 7 cable that shows more than 3 wires broken in any 1 inch [2.54 cm] length of cable.
(2) Replace any 7 x 19 cable that shows more than 6 wires broken in any 1 inch [2.54 cm] length of cable.
(3) Replace any cables where individual wires appear to blend together with the adjacent wires.
(4) Replace any control cable where evidence of rust or corrosion is found.
(5) Replace any control cable that has one broken strand of wires.

C. Trim System Operational Check *(Aircraft 35-002 thru 35-505, 36-002 thru 36-053 not modified per AAK 83-2.)*

NOTE: Refer to Chapter 5 for the current inspection interval for the Trim System Operational Check.

(1) Set the Battery Switches on.
(2) Set the PITCH TRIM switch to SEC.
(3) Hold the NOSE DN-OFF-NOSE UP switch to NOSE UP, and then to NOSE DN.
(4) Check for horizontal stabilizer movement in both directions.

NOTE: Stabilizer movement will be at approximately half the rate of the primary trim system.

(5) Set the Pilot's Control Wheel Trim Switch to NOSE UP, and then to NOSE DN. Trim motion must not occur.
(6) Set the PITCH TRIM switch to OFF.
(7) Actuate the Pilot's and Copilot's Control Wheel and Trim Arming Switches and the pedestal NOSE DN-OFF-NOSE UP switch. Trim must not occur.
(8) Set the PITCH TRIM switch to PRI.
(9) Set the NOSE DN-OFF-NOSE UP switch, on the pedestal, to NOSE UP, and then NOSE DN. Trim motion must not occur.
(10) Set the Pilot's Control Wheel Trim Switch, without pushing the arming button, move the switch to LWD, RWD, NOSE UP, and NOSE DN. Trim motion must not occur.
(11) Push the arming button. Trim motion must not occur.
(12) Push the arming button, and set the Pilot's Control Wheel Trim Switch to LWD, RWD, NOSE UP, and NOSE DN. Proper trim motion must occur.
(13) Push the arming button and move the switch to LWD, RWD, NOSE UP, and NOSE DN. Proper trim motion must occur.
(14) Repeat steps (1) thru (13) for the Copilot's Control Wheel Trim Switch.

(15) Trim by positioning the Copilot's Control Wheel Trim Switch, then trim in the opposite direction using the Pilot's Control Wheel Trim Switch. The Pilot's Control Wheel Trim Switch must override Copilot's Control Wheel Trim Switch. Repeat for each trim position.
(16) Set the Pilot's Control Wheel Trim Switch to NOSE UP, and then push the Control Wheel Master Switch. Trim motion must stop while the Control Wheel Master Switch is held.
(17) Repeat step (16) with the Pilot's Control Wheel Trim Switch set to NOSE DN. Trim motion must stop as before.
(18) Repeat steps (16) and (17) using the Copilot's Control Wheel Trim Switch and Control Wheel Master Switch.
(19) Move each half of the Yaw Trim Switch separately to NOSE LEFT and NOSE RIGHT. Trim motion must not occur.
(20) Move both halves of the Yaw Trim Switch simultaneously to NOSE LEFT and NOSE RIGHT. Proper trim motion must occur.
(21) Set all axes of the trim for takeoff and check that the T.O. TRIM Annunciator is not on.
(22) Set the Battery Switches to off.

D. Trim Systems Operational Check *(Aircraft 35-506 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per AAK 83-2.)*

NOTE: Refer to Chapter 5 for the current inspection interval for the Trim System Operational Check.

(1) Set the Battery Switches on.
(2) Make sure that the flaps are retracted.
(3) Set the Pitch Trim Selector Switch, labeled PITCH TRIM-PRI-OFF-SEC, to PRI.
(4) Trim the aircraft to nose up, then to nose down. Check for horizontal stabilizer movement in both directions. Movement must be accompanied by an audio clicker (cricket).
(5) Set the Pitch Trim Selector Switch to SEC.
(6) Set the Secondary Pitch Trim Switch to NOSE UP, then NOSE DN. Check for horizontal stabilizer movement in both directions. Movement must be accompanied by an audio clicker (cricket).
(7) Set the Secondary Pitch Trim Switch to NOSE UP. While trimming, push the Pilot's Control Wheel Master Switch. Trimming motion must stop when the Control Wheel Master Switch is pushed.
(8) Set the Secondary Pitch Trim Switch to NOSE DN. While trimming, push the Pilot's Control Wheel Master Switch. Trimming motion must stop when the Control Wheel Master Switch is pushed.
(9) Using the Copilot's Control Wheel Master Switch, repeat steps (7) and (8).
(10) Set the Pilot's Control Wheel Trim Switch (with Trim Arming push button pushed) to NOSE UP and then NOSE DN. Trim motion must not occur.
(11) Using the Copilot's Control Wheel Trim Switch, repeat step (10).
(12) Set the Pitch Trim Selector Switch to OFF.
(13) Set the Pilot's Control Wheel Trim Switch (with Trim Arming push button pushed) to NOSE UP and NOSE DN. Trim motion must not occur.
(14) Using the Copilot's Control Wheel Trim Switch, repeat step (13).
(15) Set the Secondary Pitch Trim Switch to NOSE UP and then NOSE DN. Trim motion must not occur.
(16) Set the Pitch Trim Selector Switch to PRI.

(17) Set the Secondary Pitch Trim Switch to NOSE UP and then to NOSE DN. Trim motion must not occur.
(18) Without pushing the Pilot's Trim Arming push button, set the Pilot's Control Wheel Trim Switch to LWD, RWD, NOSE UP, and NOSE DN. Trim motion must not occur.
(19) Push the Pilot's Trim Arming push button, trim motion must not occur.
(20) Set the Pilot's Control Wheel Trim Switch to NOSE UP and then to NOSE DN, while pushing the Pilot's Trim Arming push button. Trim motion must occur and an audio clicker must sound.
(21) Set the Pilot's Control Wheel Trim Switch to LWD, then to RWD while pushing the Pilot's Trim Arming push button. Roll trim motion must occur without the audio clicker.
(22) Set the Pilot's Control Wheel Trim Switch to NOSE UP while pushing the Pilot's Trim Arming pushbutton. Observe the trimming motion and audio clicker. Push the Pilot's Control Wheel Master Switch. Observe that the trimming motion and audio clicker stop.
(23) Using the Copilot's Control Wheel Trim Switch, Copilot's Trim Arming push button, and Copilot's Control Wheel Master Switch, repeat steps (18) thru (22).
(24) Trim the aircraft using the Copilot's Control Wheel Trim Switch, then trim the aircraft in the opposite direction using the Pilot's Control Wheel Trim Switch. Observe that the Pilot's Control Wheel Trim Switch overrides the Copilot's Control Wheel Trim Switch.
(25) Repeat step (24) for each trim position.
(26) Set the Test Selector Switch to TRIM OVSP.
(27) Set the Pitch Trim Selector Switch to PRI.
(28) Trim the aircraft to NOSE UP using the Pilot's Control Wheel Trim and Trim Arming Switch.
(29) Push the Press-To-Test push button. Make sure that the PITCH TRIM annunciator on the glareshield comes on.
(30) Release the Press-To-Test push button and stop trimming the aircraft.
(31) Set the Test Selector Switch to TRIM MON and push the Press-To-Test push button. Make sure that the PITCH TRIM annunciator comes on.
(32) Release the Press-To-Test push button and set the Test Selector Switch to OFF.
(33) Perform steps (3) thru (5). Make sure that the pitch trim system operates properly.
(34) Trim all the axes for takeoff.
(35) Make sure that the T.O. light goes off.
(36) Set the Battery Switches to off.

3. Repairs

A. Control Cable Pressure Seal Replacement (See Figure 210.)

NOTE: Six control cables (elevator, aileron and rudder) pass through frame 15 and are pressure sealed with a gasket held in place by a two-piece phenolic block. The gasket is designed to eliminate the need for control cable removal when the gaskets are replaced. If cabin pressure leakage becomes excessive or noise develops from air escaping around the cables, the gasket must be replaced.

(1) Disconnect electrical power from the aircraft.
(2) Remove the carpet to gain access to the aft center floorboard.
(3) Remove the aft center floorboard.
(4) Remove the forward keelbeam access cover.

(5) Remove the parts that attach the phenolic blocks to frame 15.
(6) Remove phenolic block and gasket.
(7) Set the new gasket at its correct location on frame 15 over the control cables.

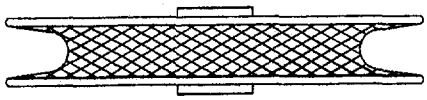
NOTE: The gasket is cut from the hole to the outer edge in three places. The cuts allow the gasket to slip over the cables and form a seal around them.

(8) Make sure that the cuts in the gasket are correctly positioned after installation.
(9) Install the phenolic block halves at their correct location on the cables.
(10) Install the parts that attach the phenolic block halves to frame 15.

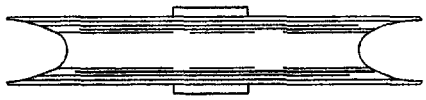
CAUTION: TIGHTEN THE ATTACHING PARTS HAND TIGHT ONLY. DO NOT TIGHTEN TO THE POINT OF DISTORTING THE SEAL.

DO NOT APPLY ANY SEALANT TO THE GASKET.

(11) Tighten the attaching parts hand tight only.
(12) Install the forward keelbeam access cover.
(13) Install the aft center floorboards.
(14) Install the carpet.
(15) Connect electrical power to the aircraft.



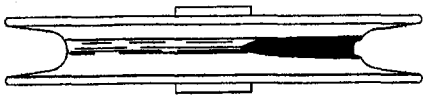
EXCESSIVE CABLE TENSION



PULLEY MISALIGNMENT

PULLEY TOO LARGE FOR CABLE

CABLE MISALIGNMENT

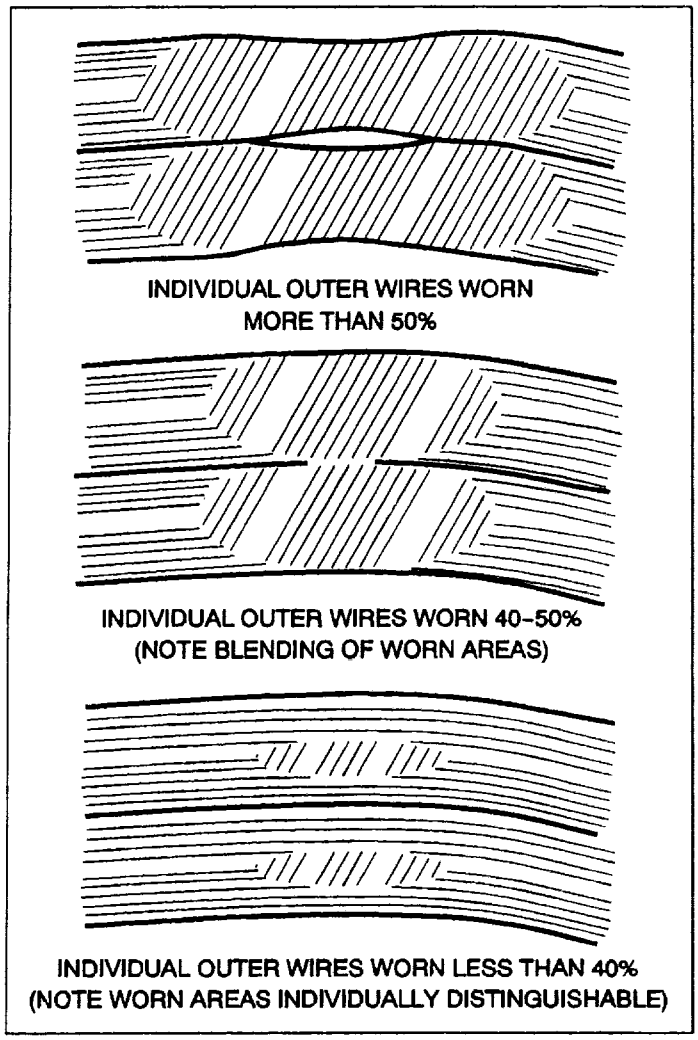


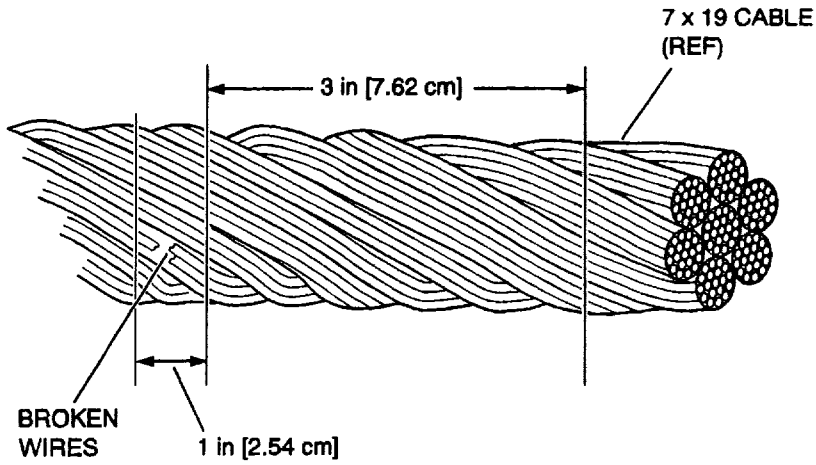
FROZEN BEARING

NORMAL CONDITION

M60-270000-207-01

Control Cable Pulley Wear Patterns
Figure 208





Control Cable Damage Limits
Figure 209

4. Cleaning/Painting

A. Application of the Corrosion Preventive *(1056H)*

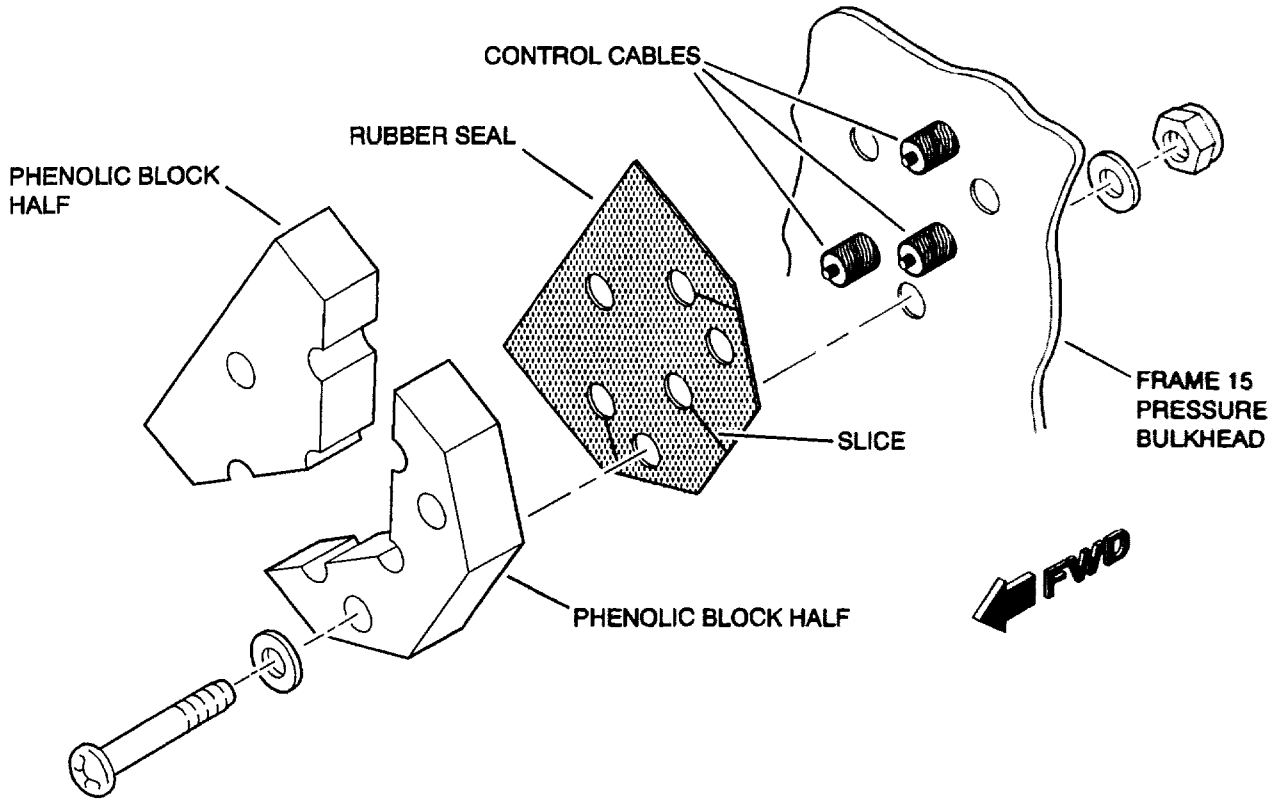
NOTE: *On Aircraft with lead-acid batteries installed*, sections of control cable assemblies near the batteries must be coated with a corrosion preventive.

(1) Get the necessary tools and equipment.

NOTE: You can use equivalent alternatives for these items:

NAME	PART NUMBER	MANUFACTURER	USE
Par-Al-Ketone		Black Bear Co. Inc. Long Island City, New York	Corrosion preventive.
Cleaning Solvent (Aliphatic Naptha, Type II)		Refer to 20-13-00.	Clean the cable.
Small Paint Brush		Commercially Available	Apply the corrosion preventive.

(2) If necessary, clean the area of the cable to be coated by wiping it with a clean cloth dampened with cleaning solvent. (Refer to 20-13-00.)
(3) Apply an even coat of corrosion protective film on the control cables in the area where they could be contaminated by the lead-acid batteries.
(4) Touch up any areas where corrosion protective film has been scrapped off, scratched, or otherwise damaged.



TYPICAL TWO PLACES

M35-270000-201-01

Control Cable Pressure Seal
Figure 210

AILERON AND TAB - DESCRIPTION AND OPERATION

1. Description

A. The aileron control system is actuated manually or automatically through the autopilot system. The aileron system consists of four subsystems; an aileron control system, an aileron trim control system, and aileron balance tab control system, and an aileron augmentation system.

B. The aileron control system provides lateral control of the aircraft. Aileron control is provided manually through the control wheels. The ailerons are balanced to a nose-heavy overbalance by lead weights in the leading edge.

C. The aileron trim tab system is an integral part of the left aileron and further increases lateral stability of the aircraft. The Trim and Trim Arming Switch, on the control wheel, controls the tab actuator. An aileron trim indicator on the cockpit center pedestal indicates trim tab position.

D. Aileron balance tabs are installed on both ailerons to decrease control wheel force. Balance tabs are controlled mechanically by push-pull tubes attached to the fixed wing surface. As the ailerons move the balance tabs move proportionally in the opposite direction.

E. The aileron augmentation system consists of the spoilers working independently to assist the ailerons. During aileron augmentation mode, the spoilers are controlled by an electrical aileron monitoring system. As one aileron moves up, its respective spoiler moves up at a 1:1 ratio with the aileron.

F. An aileron-rudder spring interconnect is installed to increase free aileron response to rudder input.

2. Component Description (See Figure 1.)

A. Aileron

(1) The ailerons are hinged at three places on the aft spar of each wing outboard of the flaps. Separate closed loop cables connect each aileron drive pulley to a common sector on the aircraft centerline. A follow-up connected to each aileron drive pulley provides output for the augmentation mode. Balance tabs are located on the trailing edge of each aileron and a trim tab is located on the inboard trailing edge of the left aileron.

(2) Small drain holes are drilled in the lower surface of the aileron just aft of the skin splice. On *Aircraft 35-274 and Subsequent except 35-278 and 36-045 and Subsequent and prior aircraft modified per AMK79-11,* drain holes and drain fairings are installed along the aileron trailing edge. Three drain holes and fairings are installed on the LH aileron; two drain holes and fairings are installed on the RH aileron.

(3) Brush seals are installed at both ends of the aileron on the wing structure. On *Aircraft 35-279 and Subsequent and 36-045 and Subsequent and prior aircraft modified per AAK 79-10,* additional brush seals are installed on the wing structure along the leading edge of the aileron. The aileron brush seals require periodic lubrication. (Refer to Chapter 12 for lubrication requirements and intervals.)

B. Aileron Trim Tab

(1) The aileron trim tab is hinged to the inboard trailing edge of the left aileron by a continuous-type hinge. The trim tab is controlled by a push-pull tube connected to a rotary-type electrical actuator located in the aileron leading edge.

C. Aileron Balance Tabs

(1) The aileron balance tabs are hinged to the trailing edge of each aileron by a continuous-type hinge. The balance tabs are controlled by push-pull tubes attached to the fixed wing surface.

D. Aileron Control Wheels

(1) A U-shaped control wheel on each control column controls the ailerons. Control wheel motion is transferred to the ailerons through a closed loop drive cable to a sector located on the fuselage centerline. Maximum control wheel movement is 110° both right and left of neutral. The Trim and Trim Arming Switch is located on the control wheel. (Refer to 27-10-05 for maintenance practices on the control wheel.)

3. Operation

A. Aileron Control System (See Figure 1.)

(1) When either aileron control wheel is rotated, the crossover cable rotates the other control wheel and the control cable rotates the aileron sector located on the fuselage centerline. The ailerons are interconnected through the aileron sector by separate closed loop cables. The cables from the aileron sector are connected to the aileron drive pulley which rotate and raise or lower the ailerons. Maximum control wheel movement is 110° both left and right of neutral. Maximum aileron movement is 18 (±1)° up and 18 (±2)° down.

B. Aileron Augmentation System (See Figure 2 this section and Figure 3 in 27-60-00.)

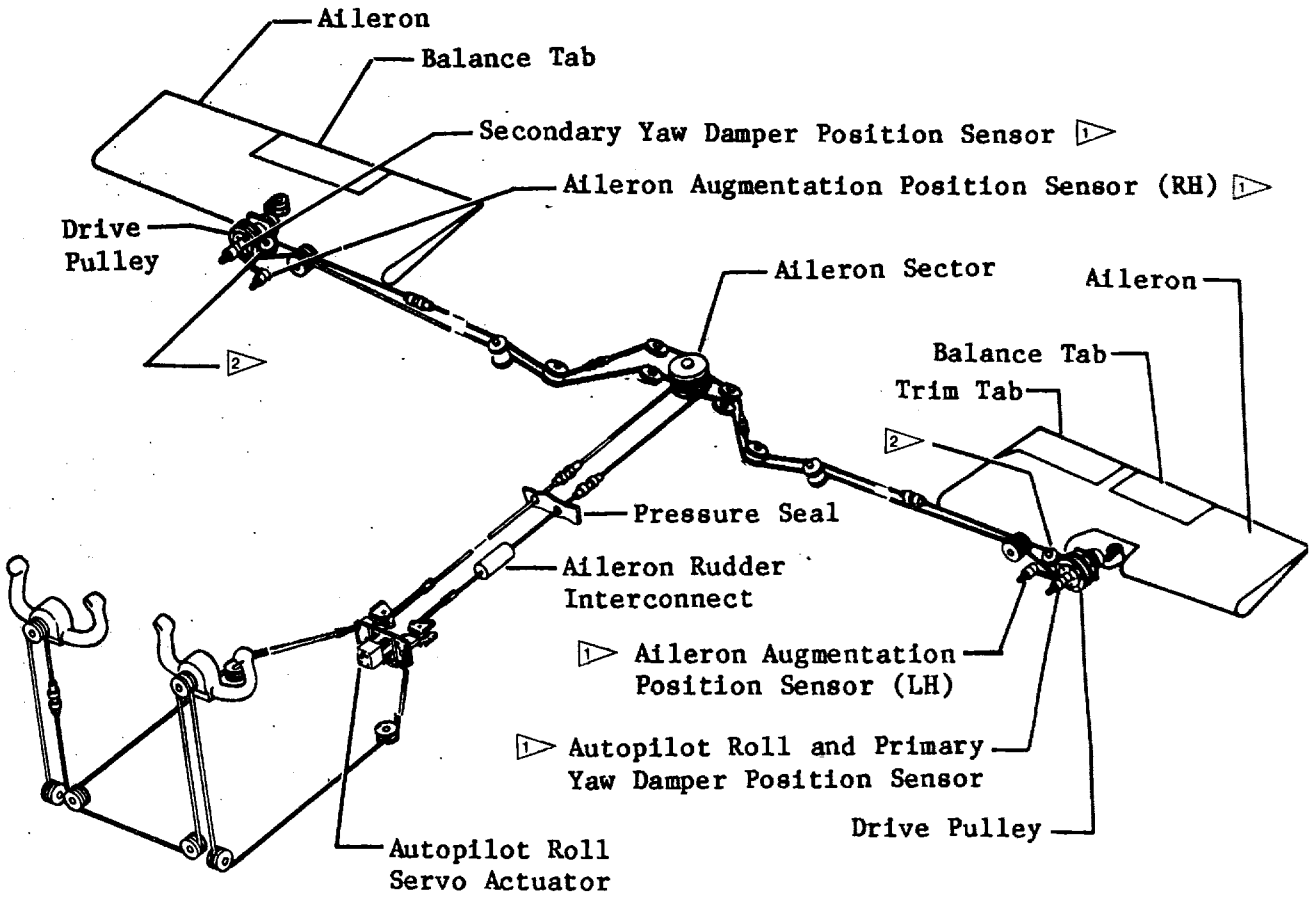
(1) The aileron augmentation mode is engaged automatically when both flaps are extended below 25° providing the spoiler switch is in the retract position. A position sensor attached to each aileron provides an input to a computer-amplifier. The computer-amplifier controls two spoiler servo valves located on a manifold assembly. When an aileron deflects up, its position sensor sends a signal to the computer-amplifier. The computer-amplifier then actuates the spoiler control valve and the spoiler servo valves which extend the applicable spoiler at a 1:1 ratio with the aileron. A monitor circuit automatically disengages the augmentation system, illuminates an AUG-AIL light (amber) and retracts the spoilers when a system malfunction occurs. A SPOILERON RESET Switch will restore system operation if the malfunction has cleared itself. Normal spoiler operation overrides the augmentation modes. The augmentation mode disengages when the flaps are retracted above 25°.

C. Aileron Trim Tab System (See Figure 3.)

(1) The Trim and Trim Arming Switch on the control wheel controls the aileron trim tab actuator in the left aileron. Depressing the Trim and Trim Arming Switch energizes the actuator to move the trim tab either up or down. A trim tab position transmitter is attached to the actuator shaft and provides input for the aileron trim tab position indicator on the cockpit center pedestal. Trim tab travel is 8 (±1)° both up and down.

D. Aileron Balance Tab System

(1) The aileron balance tabs are hinged to the aileron and connected to the fixed wing surface by push-pull tubes. As the ailerons are moved, the balance tabs move proportionately in the opposite direction. With the aileron fully up, the balance tab will be 15 (±2)° down relative to the ailerons. With the aileron fully down, the balance tab will be 15 (±2)° up relative to the ailerons.



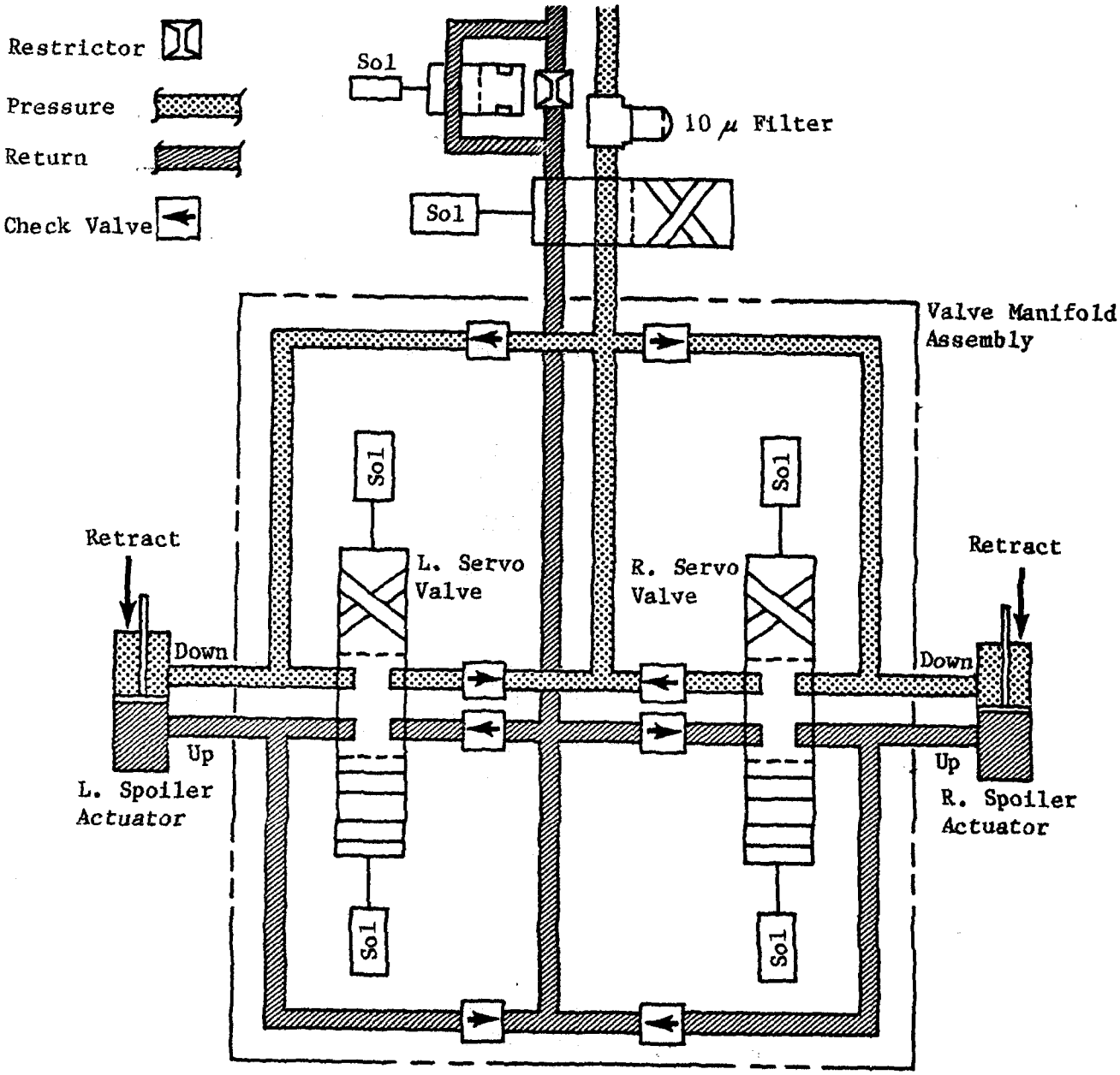
1 *Aircraft not equipped with FC-530 autopilot.*

2 This pulley effective on *Aircraft 35-116 and Subsequent; Aircraft 36-033 and Subsequent, and previous aircraft modified per AMK 77-1, "Installation of Aileron Control Cable Idler Pulleys".*

(TYPICAL)

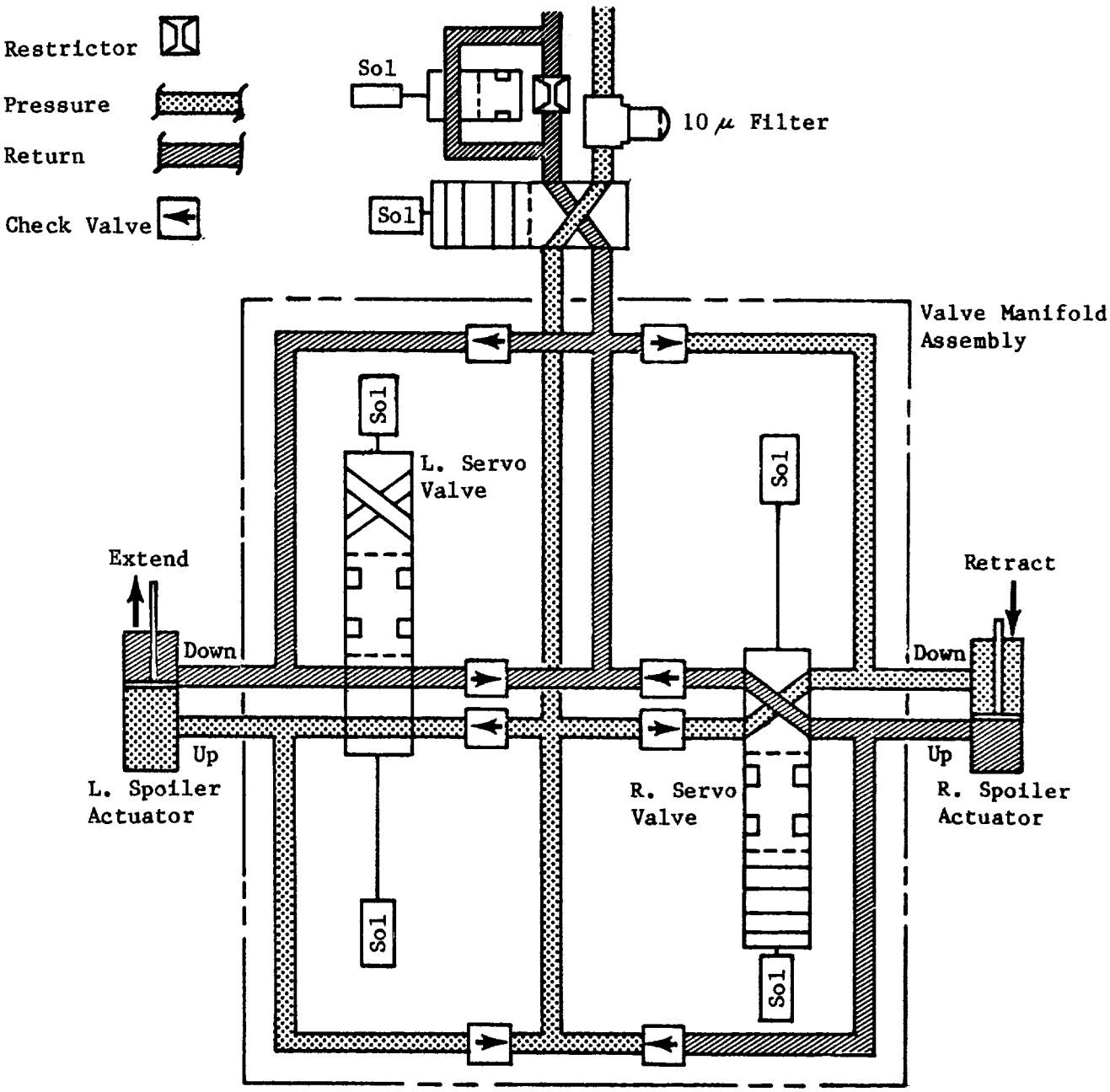
8-13B-2

Aileron Control System
Figure 1



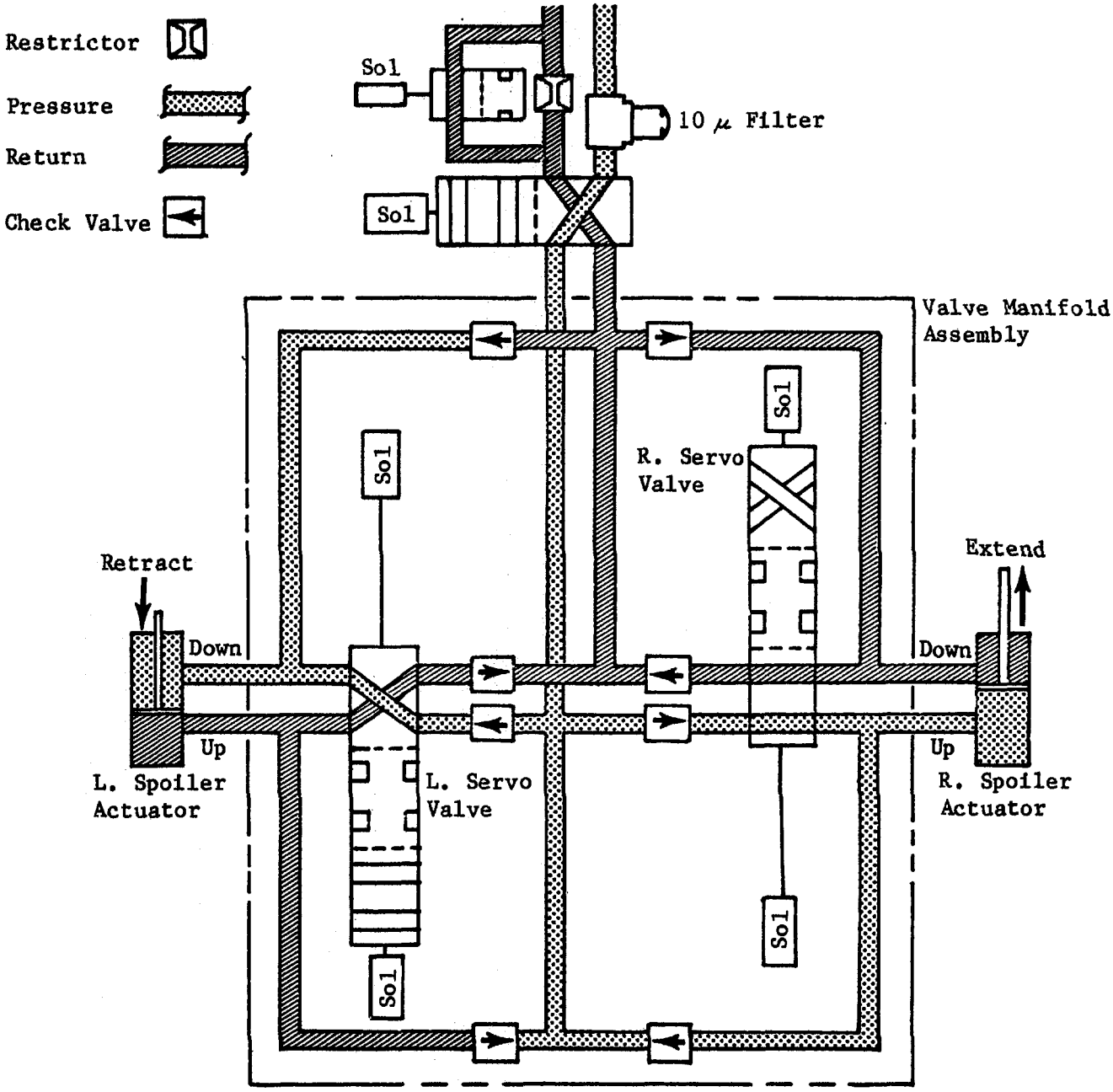
Spoilers Retracted (Spoiler Switch Set to RET-Augmentation Disengaged)

Aileron Augmentation System Hydraulic Schematic
Figure 2 (Sheet 1 of 4)



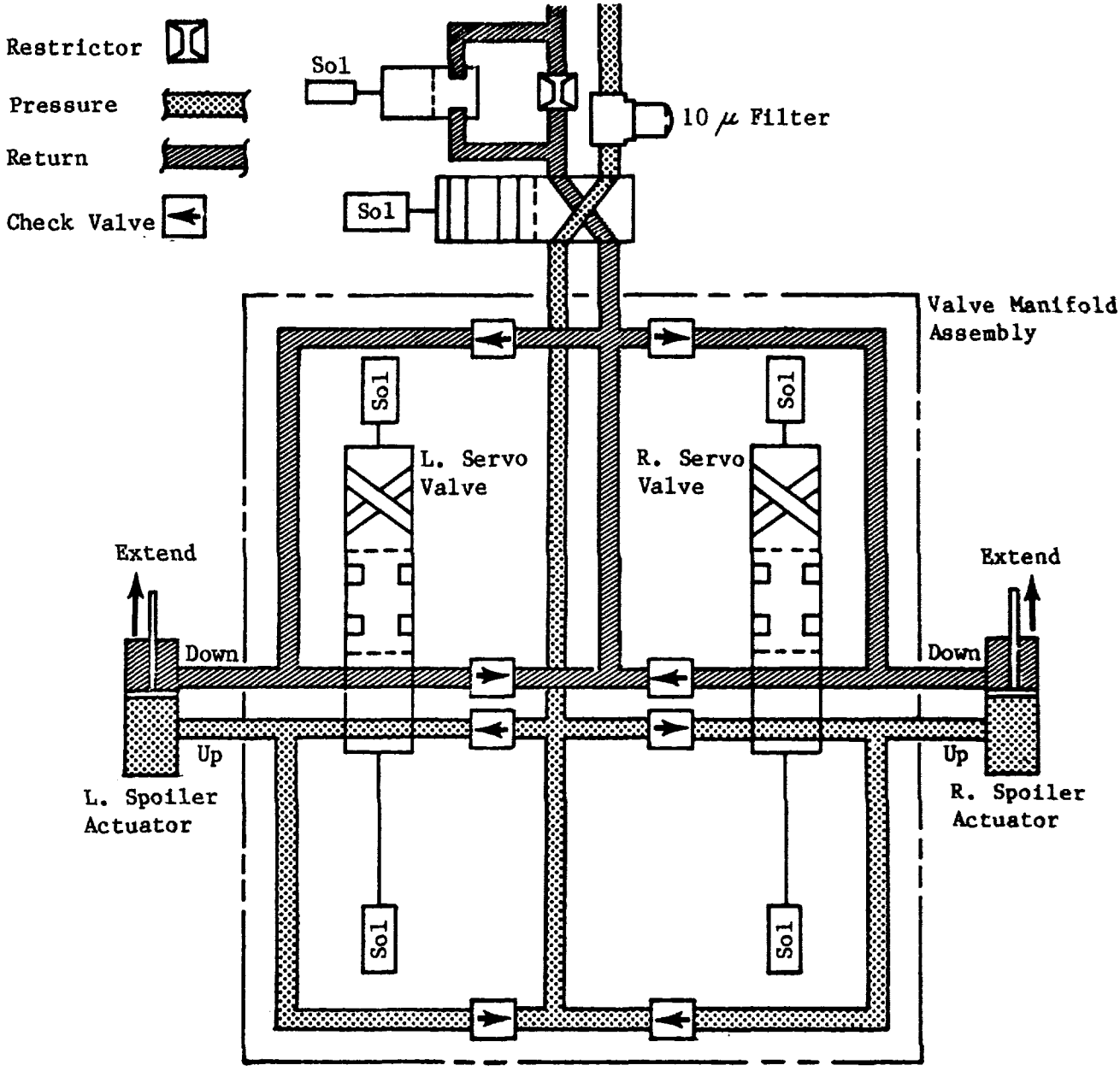
Left Aileron Augmented (Spoiler Switch Set to RET)

Aileron Augmentation System Hydraulic Schematic
Figure 2 (Sheet 2 of 4)



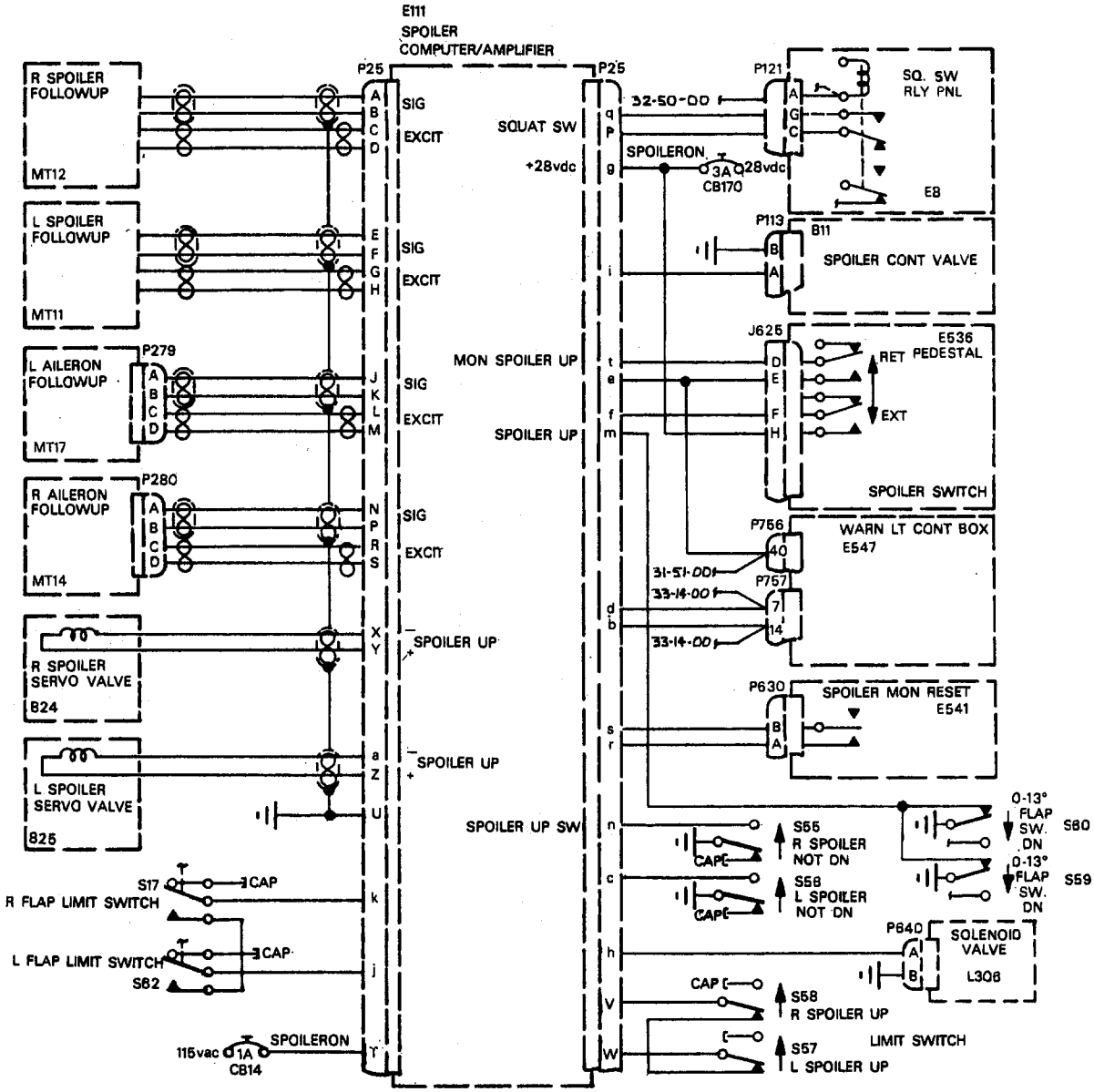
Right Aileron Augmented (Spoiler Switch Set to RET)

Aileron Augmentation System Hydraulic Schematic
Figure 2 (Sheet 3 of 4)

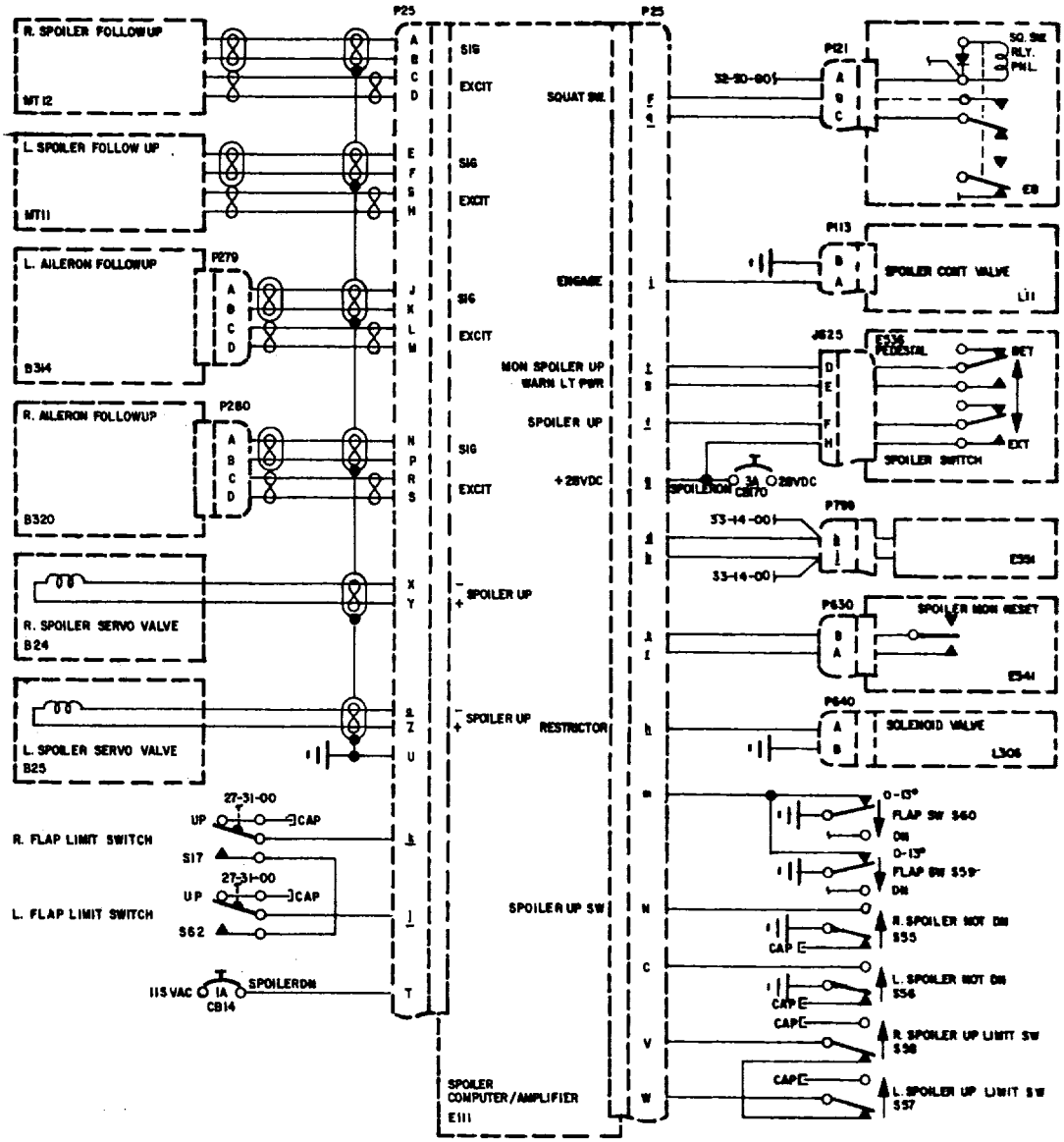


Spoilers Both Extended (Spoiler Switch Set to EXT)

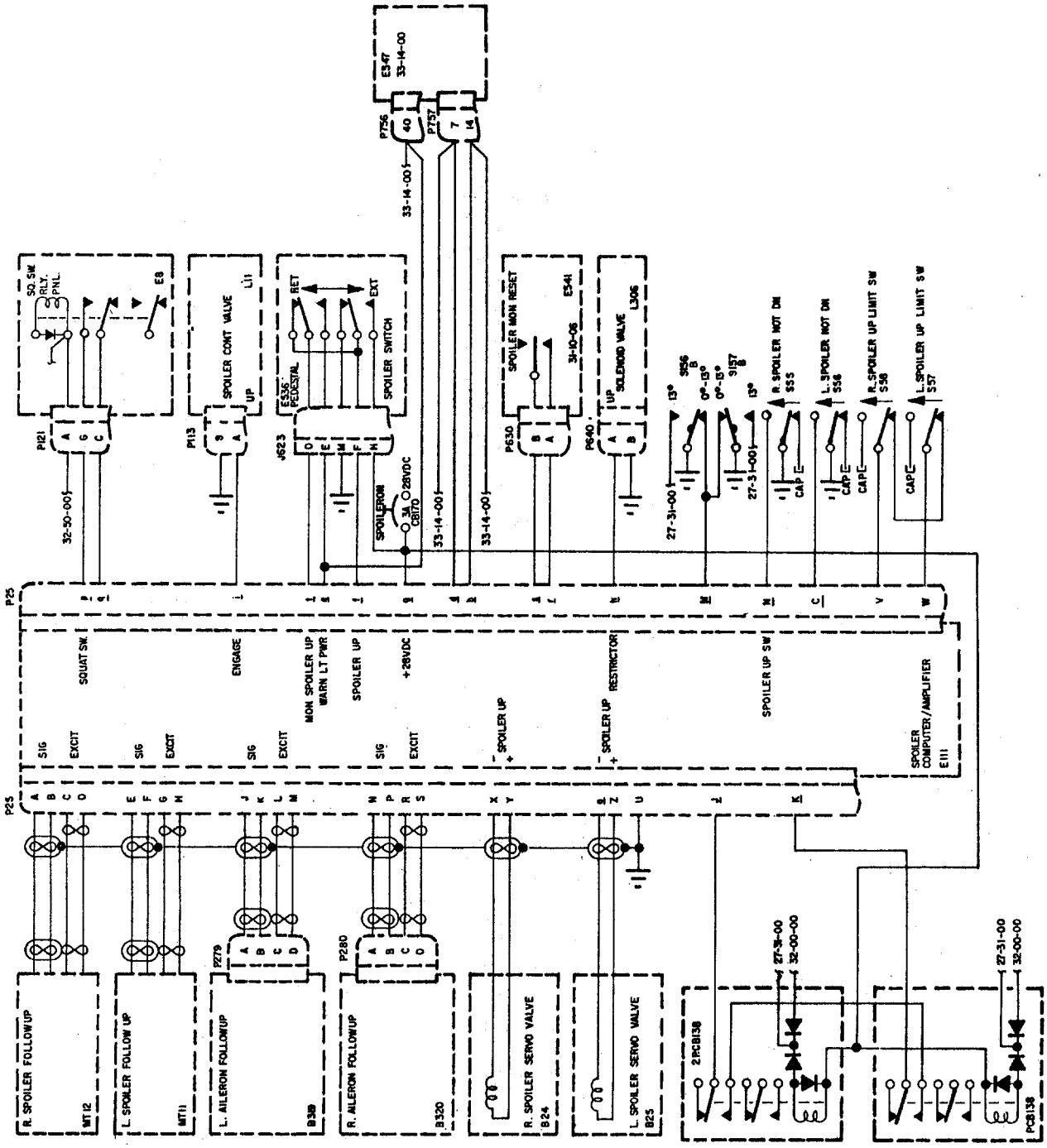
Aileron Augmentation System Hydraulic Schematic
Figure 2 (Sheet 4 of 4)



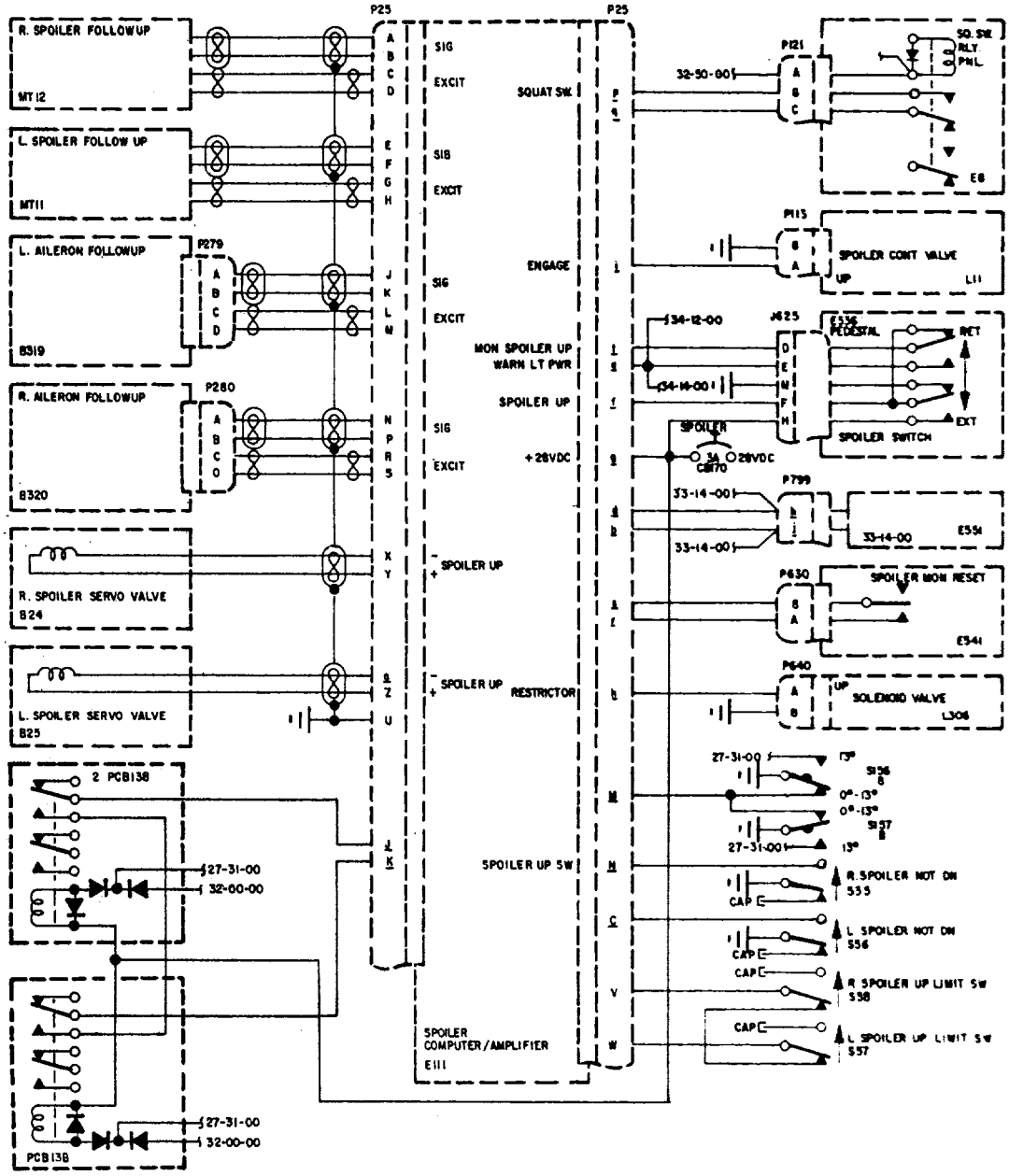
Aileron Augmentation System Electrical Control Schematic
Figure 3 (Sheet 1 of 6)



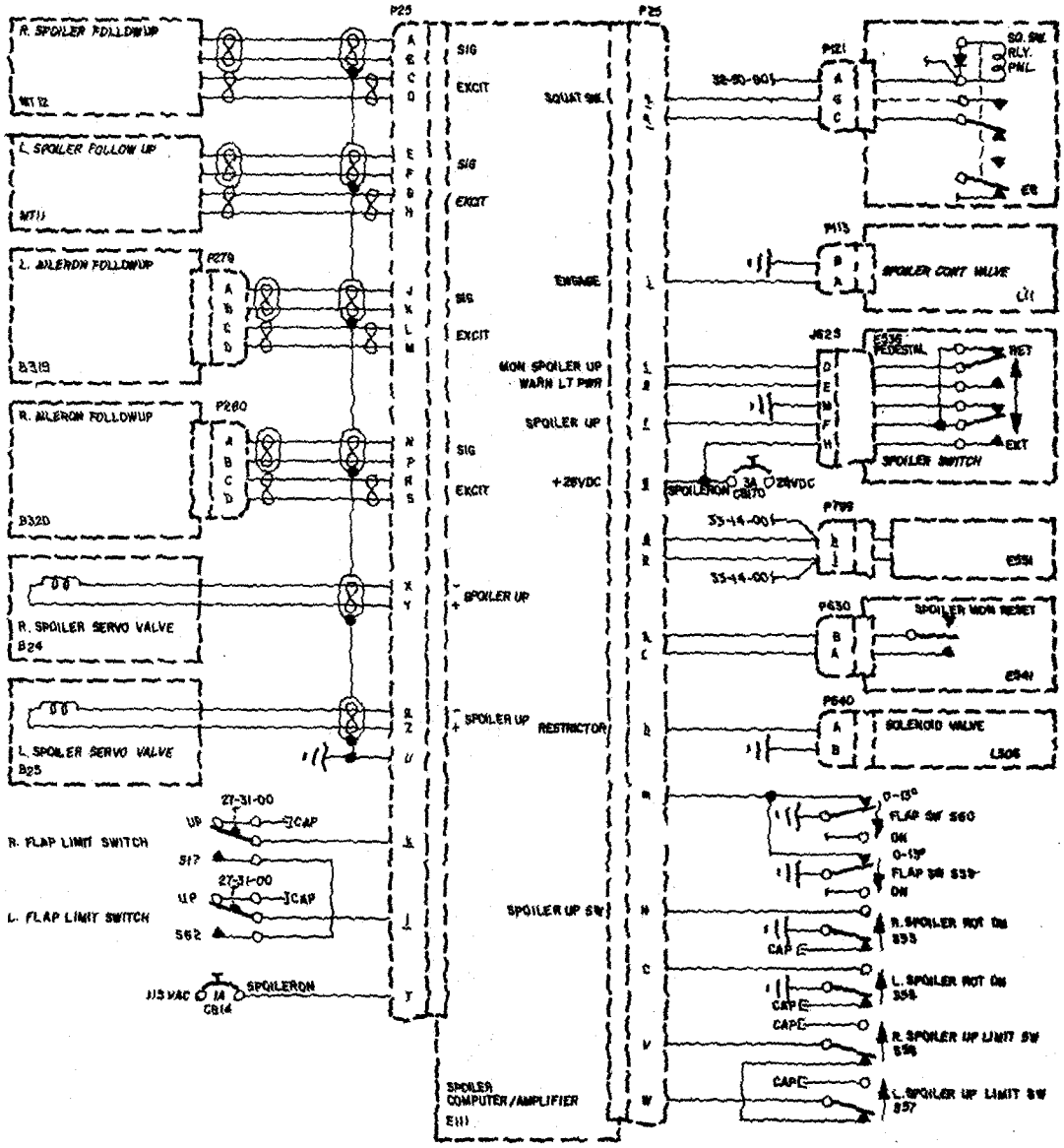
Aileron Augmentation System Electrical Control Schematic
Figure 3 (Sheet 2 of 6)



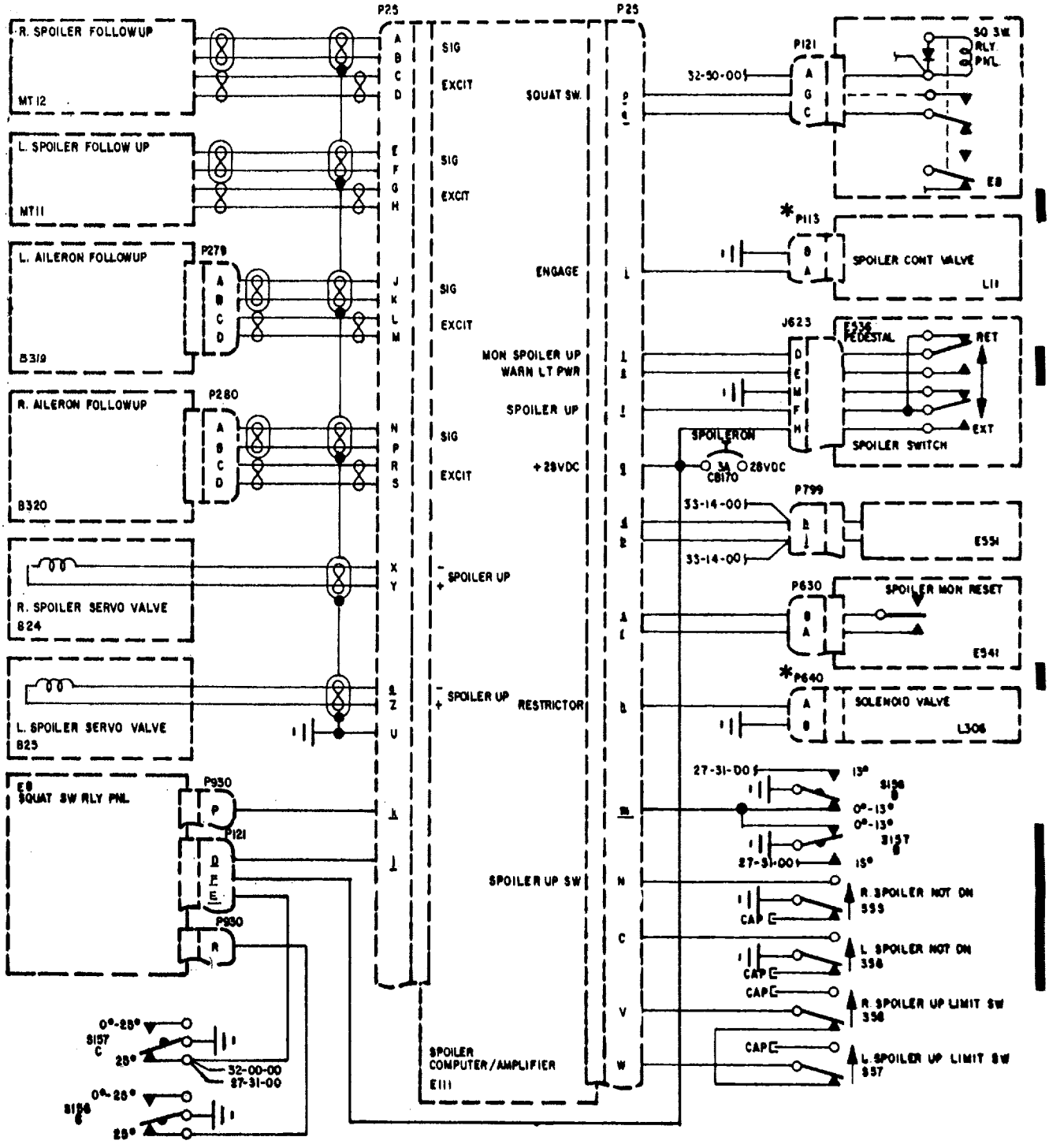
Aileron Augmentation System Electrical Control Schematic
Figure 3 (Sheet 3 of 6)



Aileron Augmentation System Electrical Control Schematic
Figure 3 (Sheet 4 of 6)

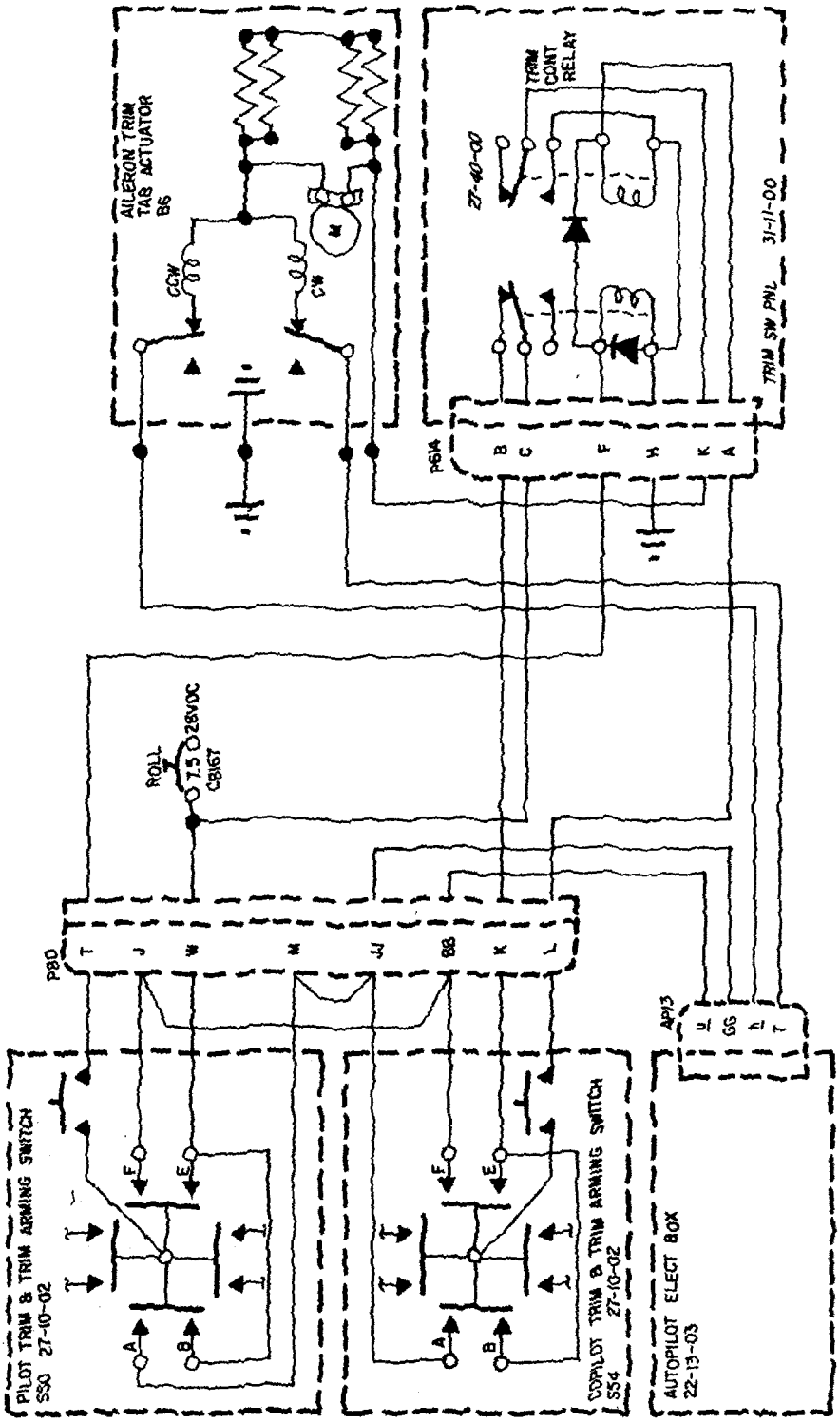


Aileron Augmentation System Electrical Control Schematic
Figure 3 (Sheet 5 of 6)

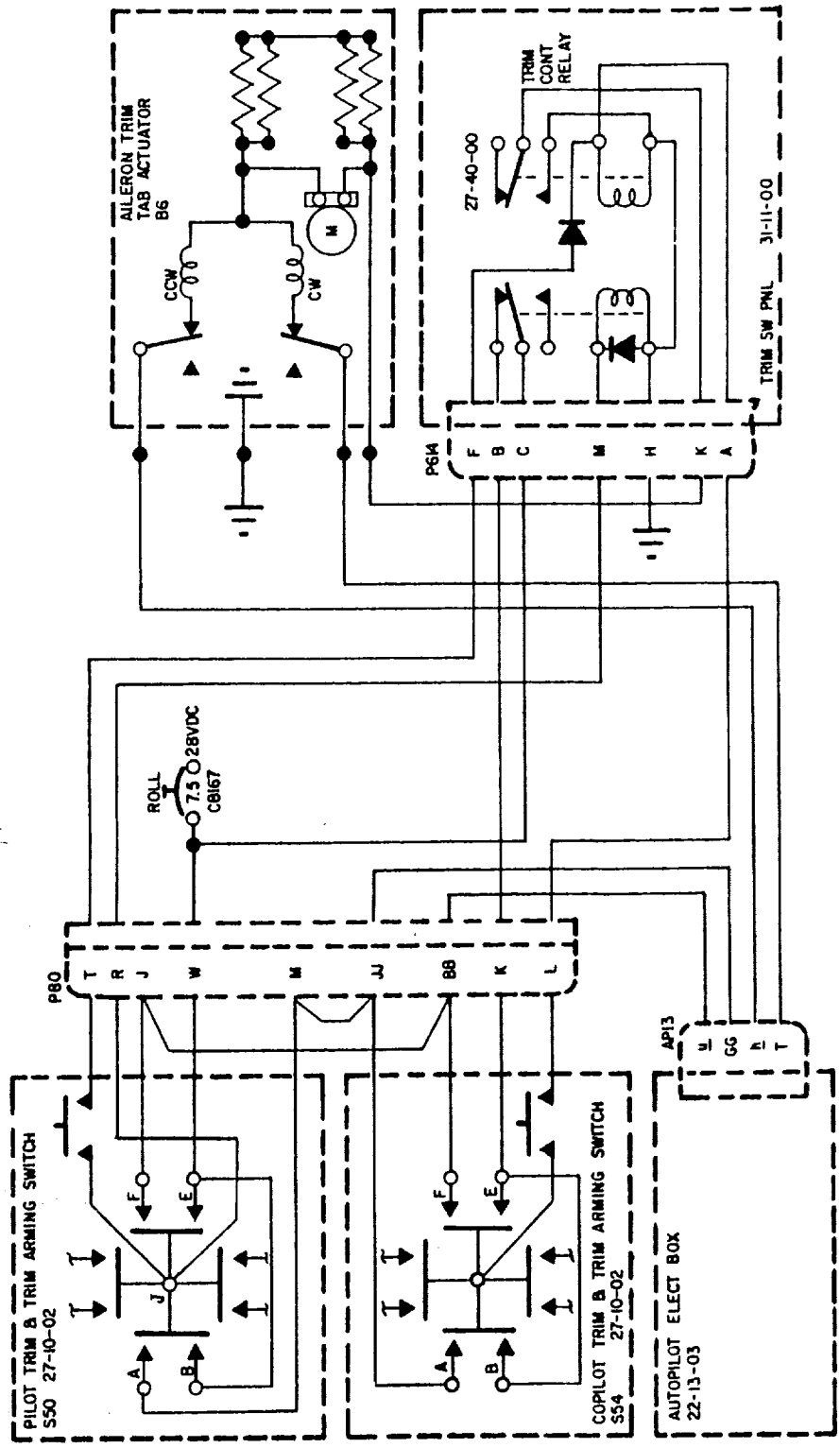


Effective Aircraft 35-117 and Subsequent and 36-033 and Subsequent, P113-B changes to P113-C, P113-A changes to P113-D, P640-A changes to P640-D, P640-B changes to P640-C.

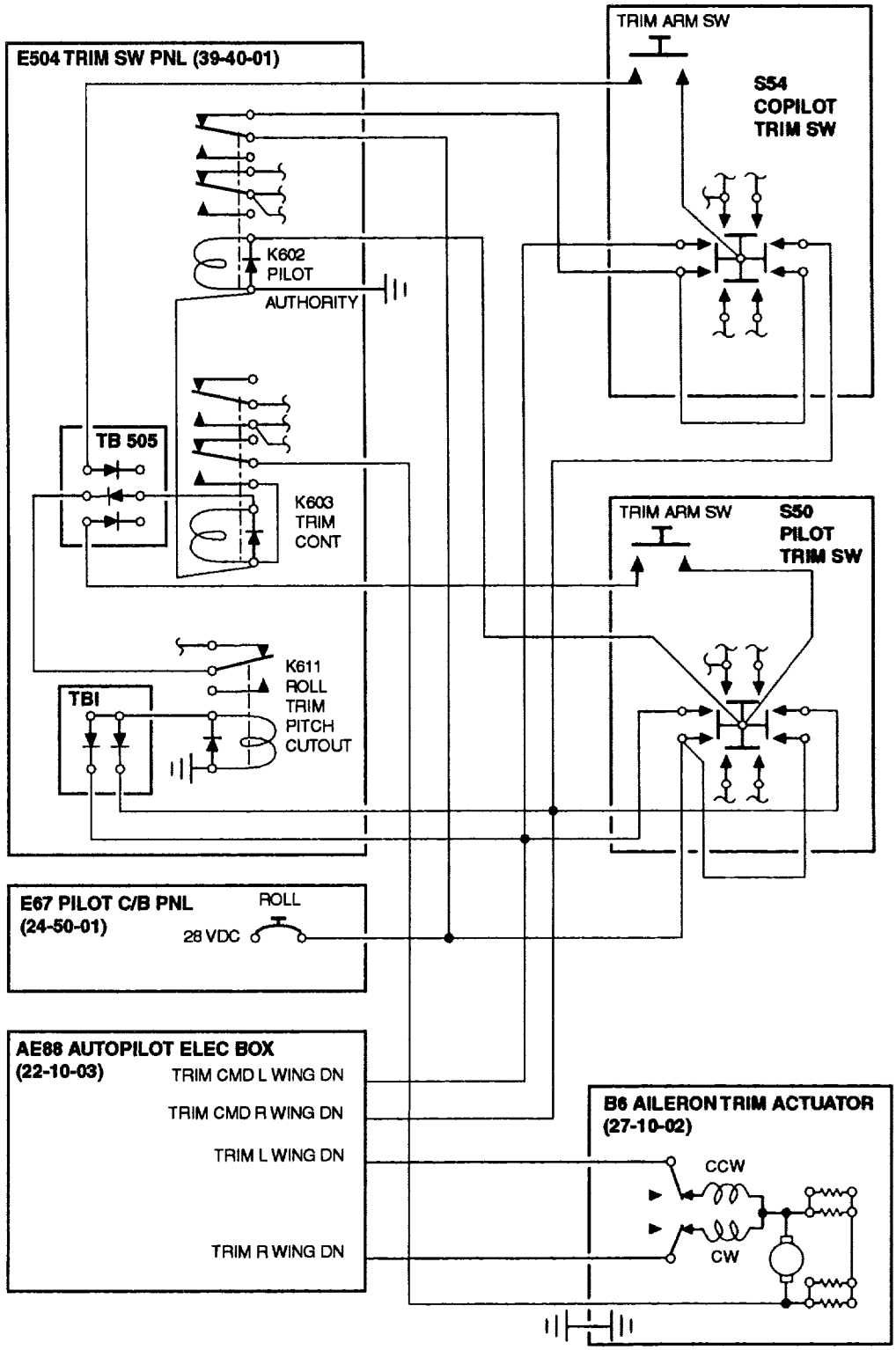
Aileron Augmentation System Electrical Control Schematic
Figure 3 (Sheet 6 of 6)



Aileron Trim Tab Electrical Control Schematic
Figure 4 (Sheet 1 of 3)



AileronTrim Tab Electrical Control Schematic
Figure 4 (Sheet 2 of 3)



Aileron Trim Tab Electrical Control Schematic
Figure 4 (Sheet 3 of 3)

AILERON AND TAB - MAINTENANCE PRACTICES

1. Description

A. The following procedures cover adjustment of the aileron, aileron-rudder interconnect, actuator trim tab and aileron balance tabs. Adjustments are typical for left and right ailerons.

B. Access for aileron rigging is gained by removal of access covers on copilot's control column, cabin center aisle, fuselage center section, and underside of wings.

2. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Contour Plate	2471003-1	Learjet Inc. Wichita, KS	Locate control surface neutral.
Contour Plate	T/N 315977	Learjet Inc. Wichita, KS	To locate control surface neutral.
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	To measure angles of control surfaces.
Digital Protractor (DP-45) (Alternate)	02475-01	Lucas & Schaebitz Pennsauken, NJ	To measure angles of control surfaces.
Tensiometer (0-150 lb [0-70 kg])		Commercially Available	Measure cable tension.
Push-Pull Spring Scale (0-10 lb [0-4.54 kg])		Commercially Available	Measure force.
Tensiometer Adapter	2471007	Learjet Inc. Wichita, KS	Measure cable tension inaccessible to normal cable tensiometer use.
Rigging Pin		Local Manufacture	Rigging controls.

3. Adjustment/Test

A. Rig Aileron Control System (See Figure 201.)

NOTE: Before making any precise adjustments, perform an operational check of the aileron system by rotating the control wheel. When the control wheel is moved to the right, the aileron on the right wing will move up and the aileron on the left wing will move down. When the control wheel is moved to the left, the aileron on the right wing will move down and the aileron on the left wing will move up. If, for any reason, a cable is removed and subsequently reattached, this check shall be repeated.

(1) Remove access covers from copilot's control column, cabin center aisle, fuselage center section, and underside of wings.
(2) Disconnect aileron autopilot servo cable.
(3) Center control wheels by laying a straightedge across the top of both control wheels. Clamp straightedge to control wheels in a suitable manner.

NOTE: Rigging pin is fabricated from 3/16 inch (4.76 millimeter) round steel stock. Make 90° bend, approximately 8-1/2 inches (21 centimeters) from end of 12 inch (30.5 centimeter) piece of stock. Paint short end of piece red (for visibility).

(4) Insert rigging pin in aileron sector.

NOTE: On *Aircraft 35-108 and Subsequent and 36-032 and Subsequent*, a hole is provided in the keelbeam access cover approximately 6 inches forward of frame 22. This hole allows the rigging pin to be inserted in the aileron sector without removal of the access panel.

(5) Attach contour plate as follows:
(a) Using contour plate P/N 2471003-1, attach to wings at WS 157 LH and RH.

NOTE: Two (2) contour plates may be used as long as part numbers are the same.

If contour plate P/N 2471003-1 comes in contact with balance tabs they must be faired and then returned to original configuration.

(b) Using contour plate T/N 315977, attach to wings at WS 181 LH and RH.

CAUTION: CONTOUR PLATE MATING SURFACES SHALL BE FREE OF ANY TAPE OR PROTECTIVE COVERING THAT MAY BE INSTALLED TO PROTECT THE AIRCRAFT PAINTED SURFACE.

USE ONE CONTOUR PLATE FOR BOTH AILERONS. THIS ENSURES THAT AILERON DROOP NEUTRAL POSITIONS ARE SYMMETRICAL.

(6) Adjust appropriate turnbuckles until both ailerons are synchronous at the centerline of trailing edge as follows:
(a) If using contour plate P/N 2471003-1, synchronize ailerons to a droop neutral position of 0.50 (±0.06) inch (12.7 [±1.5] mm) below the contour plate to centerline of aileron trailing edge.
(b) If using contour plate T/N 315977, synchronize ailerons to a droop neutral position of 0.45 (±0.05) inch (11.4 [±1.3] mm) below the contour plate to centerline of aileron trailing edge.

NOTE: The threads at opposite ends of the turnbuckle shall be started at the same time so thread engagement on both ends will be approximately equal.

The final adjusted turnbuckle combination shall have no more than three threads of the terminal, fork, or eye exposed outside the barrel of the turnbuckle.

Droop neutral position is measured from lower edge of contour plate to centerline of aileron trailing edge.

(7) Adjust aileron control cable tension in accordance with procedures outlined in 3.E.

NOTE: When cable turnbuckles are loosened or tightened to adjust cable tension, recheck aileron synchronization. If necessary, readjust and synchronize ailerons. Repeat these procedures alternately until ailerons are synchronized and correct cable tension is obtained.

(8) Attach aileron autopilot roll servo cable to aileron control system cables.
(9) Check capstan clutch setting. (Refer to 22-00-00.)
(10) Adjust autopilot roll servo cable tension to 25 (±5) pounds (11.34 [±2.27] kilograms) at 75°F (24° C). (Refer to 27-00-00 for cable tension temperature-load correction table.)

NOTE: When autopilot roll servo cable tension is adjusted, primary aileron control cable aft of attach point will be 100 (±10) pounds (45.36 [±4.54] kilograms) and primary cable tension forward of attach point will be 75 (±10) pounds (34.02 [±4.54] kilograms).

(11) Check tension on aileron control system cables aft of servo cable attachment. Tension should be 100 (±10) pounds (45.36 [±4.54] kilograms).
(12) Safety turnbuckles and remove rigging pin.
(13) Attach clinometers to ailerons to measure travel.
(14) Adjust the up-stops for both ailerons to obtain 18° (±1°) up travel. On *Aircraft 35-040 and Subsequent and 36-017 and Subsequent and aircraft modified by SB 35/36-27-3,* safety wire stop bolts.
(15) Adjust the down-stops for both ailerons to obtain 18° (±2°) down travel. The down stops should be set to match the up-stops on the opposite aileron. On *Aircraft 35-040 and Subsequent and 36-017 and Subsequent and aircraft modified by SB 35/36-27-3,* safety wire stop bolts.

NOTE: Travel is measured from "aileron neutral" position. (Refer to step 3.A.(6).) If an aileron will achieve the required travel in only one direction, it indicates that the yoke is not centered on the hinge line. The shims between the yoke bearings and the center hinge casting should be changed as necessary to obtain the correct travel.

(16) Check that the fixed stops in the control column heads do not contact until after the primary stops on the aileron drive pulley are reached.
(17) Operate ailerons several times by moving the control wheel and check for freedom of movement.
(18) Perform the aileron control system friction test as follows (with autopilot not engaged):
  (a) With the aileron and rudder systems centered, place a push-pull spring scale (accurate within 1/2 pound [0.227 kilogram]) on the horizontal arm of the pilot's control wheel 4.5 inches (6.35 centimeters) from the vertical centerline of the pilot's control wheel hub.
  (b) Slowly apply a downward force on the spring scale.
  (c) Read the force applied on the spring scale when the control wheel starts to move. The maximum allowable force is 8 pounds (1.8 kilograms).
  (d) If the applied force is more than 8 pounds (1.8 kilograms), check for proper cable tensions; check pulleys and sector for freedom of movement and alignment; visually inspect system for binding; check system for proper lubrication; and ensure that the aileron and rudder control systems are properly rigged.
  (e) After all discrepancies have been corrected, perform steps (a) thru (c) and ensure that the applied force is less than 8 pounds (1.8 kilograms).
(19) Install all floorboards and access covers.
(20) Rig spoilers. (Refer to 27-60-00.)

B. Rig Aileron-Rudder Interconnect (See Figure 202.)
(1) Remove floorboards necessary to gain access to aileron rudder interconnect.
(2) Adjust nuts on rod of spring cartridge so they contact the spring followers and the spring followers contact the retainers simultaneously on both ends. Maximum end play shall not exceed 0.010 inch (0.254 millimeter).

(3) With both the ailerons and the rudder in neutral, adjust the interconnect cable assembly to 25 (±5) pounds (11.34 [±2.27] kilograms) at 75°F (24°C). (Refer to 27-00-00 for cable tension temperature-load correction.)
(4) Operate rudder and aileron systems through full travel in all combinations and check for full freedom of movement. Left rudder should produce left aileron and right rudder should produce right aileron.
(5) Install floorboards.

C. Rig Aileron Balance Tab (See Figure 203.)

NOTE: Before rigging the aileron balance tab, the primary aileron system must be rigged per step 3.A.

(1) Attach clinometer to aileron balance tab.
(2) Alternately move each aileron to the droop neutral position. (Refer to steps 3.A.(6)(a) and (b).)
(3) Adjust tab linkage to the flight neutral position.

NOTE: If opposite wings balance tab is in the faired position, the balance tab must be adjusted to the faired position. If opposite wings balance tab is positioned up, the balance tab must be positioned down the same amount or if opposite wings tab is positioned down, the balance tab must be positioned the same amount up.

EXAMPLE: If aileron trim tab position is 1 mark left wing down, then lengthen right wing balance tab pushrods 1 turn and shorten left wing balance tab pushrods 1 turn.

The aircraft logbook will indicate if the faired position is acceptable or the number of turns and direction made to each sides pushrods to establish a flight neutral position.

(4) Balance tab travel shall be measured to fall within the following :

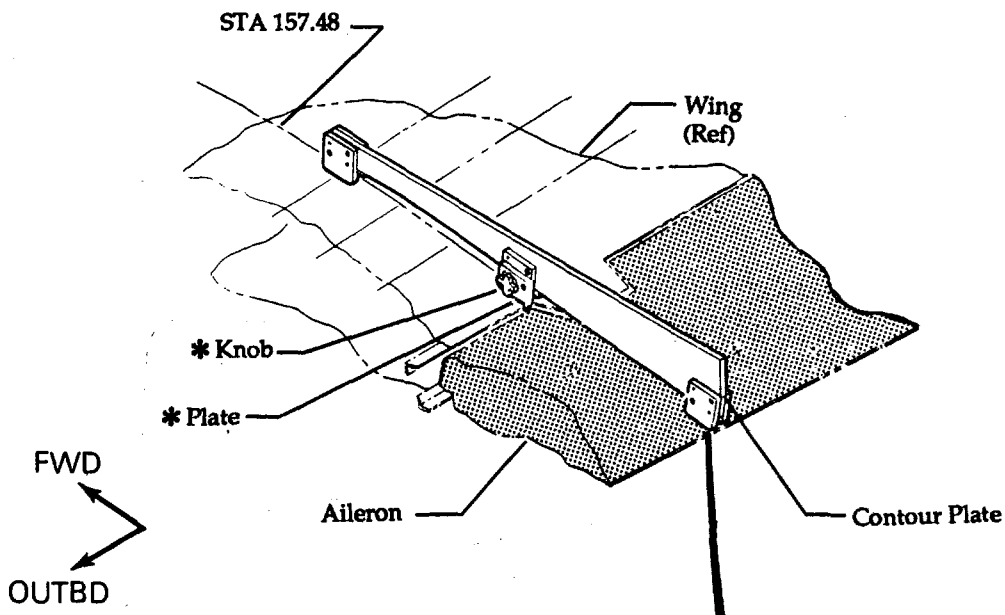
	BALANCE TAB TRAVEL RELATIVE TO AILERON
Aileron Full Down	15 (±2)°
Aileron Full Up	15 (±2)°

(5) Remove clinometer from balance tab.

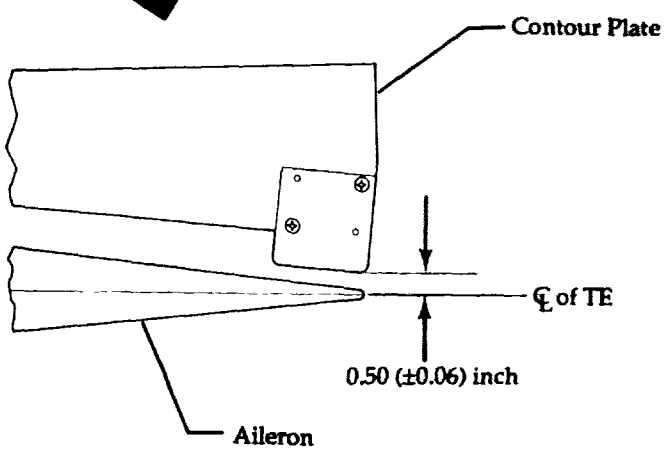
D. Rig Aileron Trim Tab
(1) Remove trim tab actuator access cover on aileron leading edge.
(2) Actuate trim tab to streamline neutral position using the Trim and Trim Arming Switch on the control wheel.
(3) Attach clinometer to aileron trim tab.
(4) Check trim tab travel for 8 (±1)° both up and down.

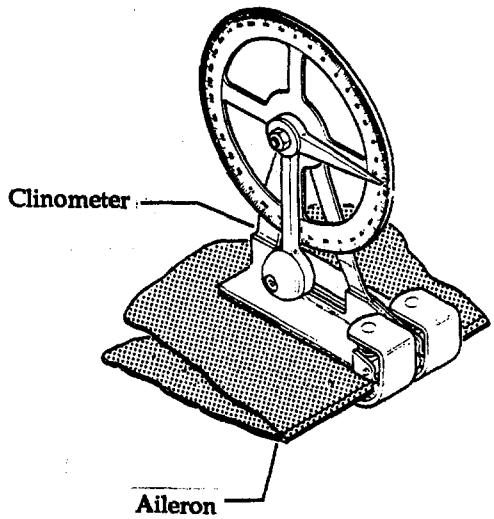
NOTE: If limit switch adjustment is necessary, refer to Rotary Actuator Repair Manual, Pub. No. TM6314, for adjustment procedures.

(5) Cycle trim tab several times, checking for proper operation and correlation between aileron trim tab and aileron trim indicator on the cockpit center pedestal.
(6) Install trim tab actuator access cover.



* **INSTALL KNOB AND PLATE ON OUTBOARD SIDE OF CONTOUR PLATE.**





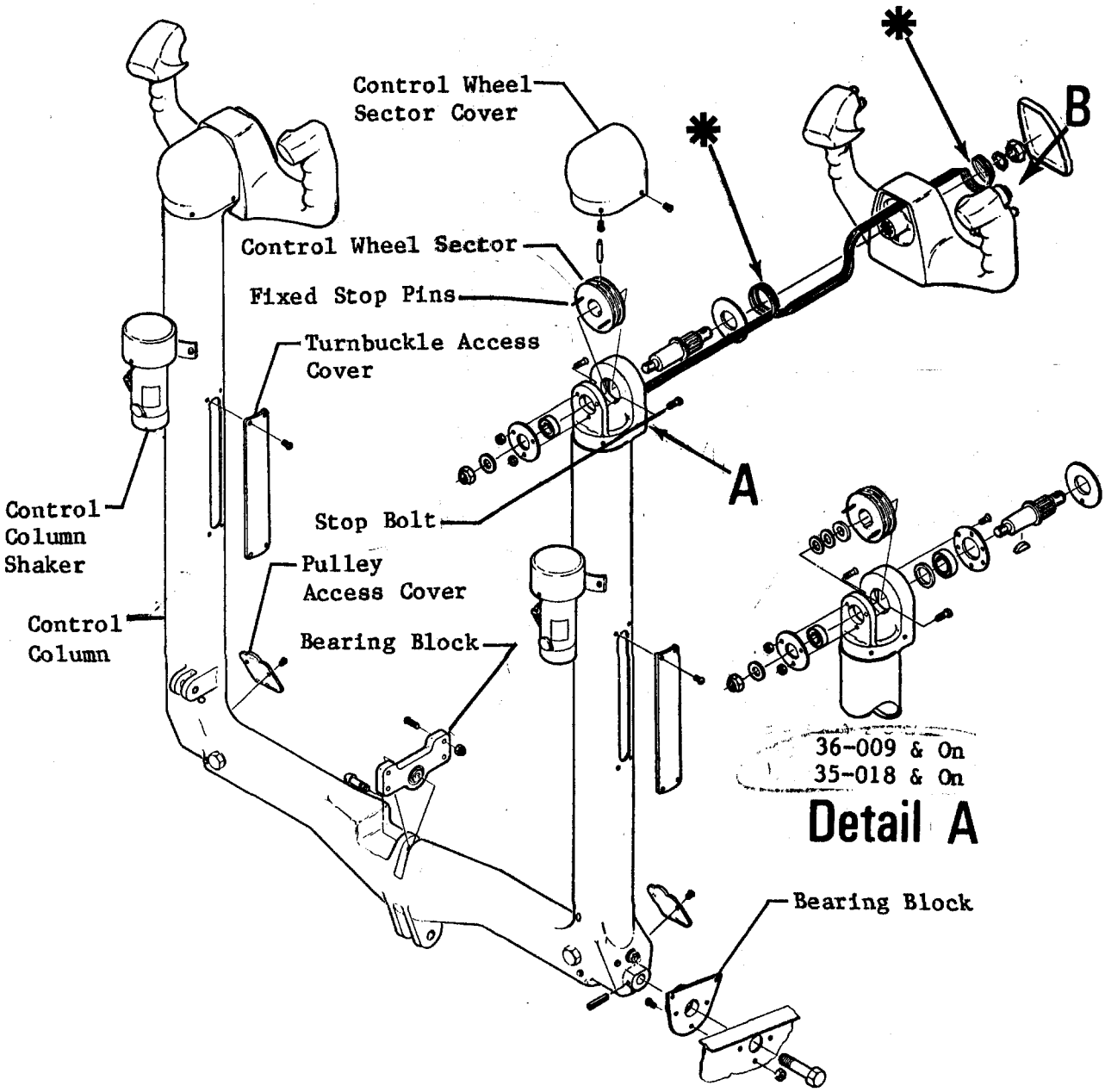
14-235A

Aileron Control System Rigging
Figure 201 (Sheet 1 of 5)

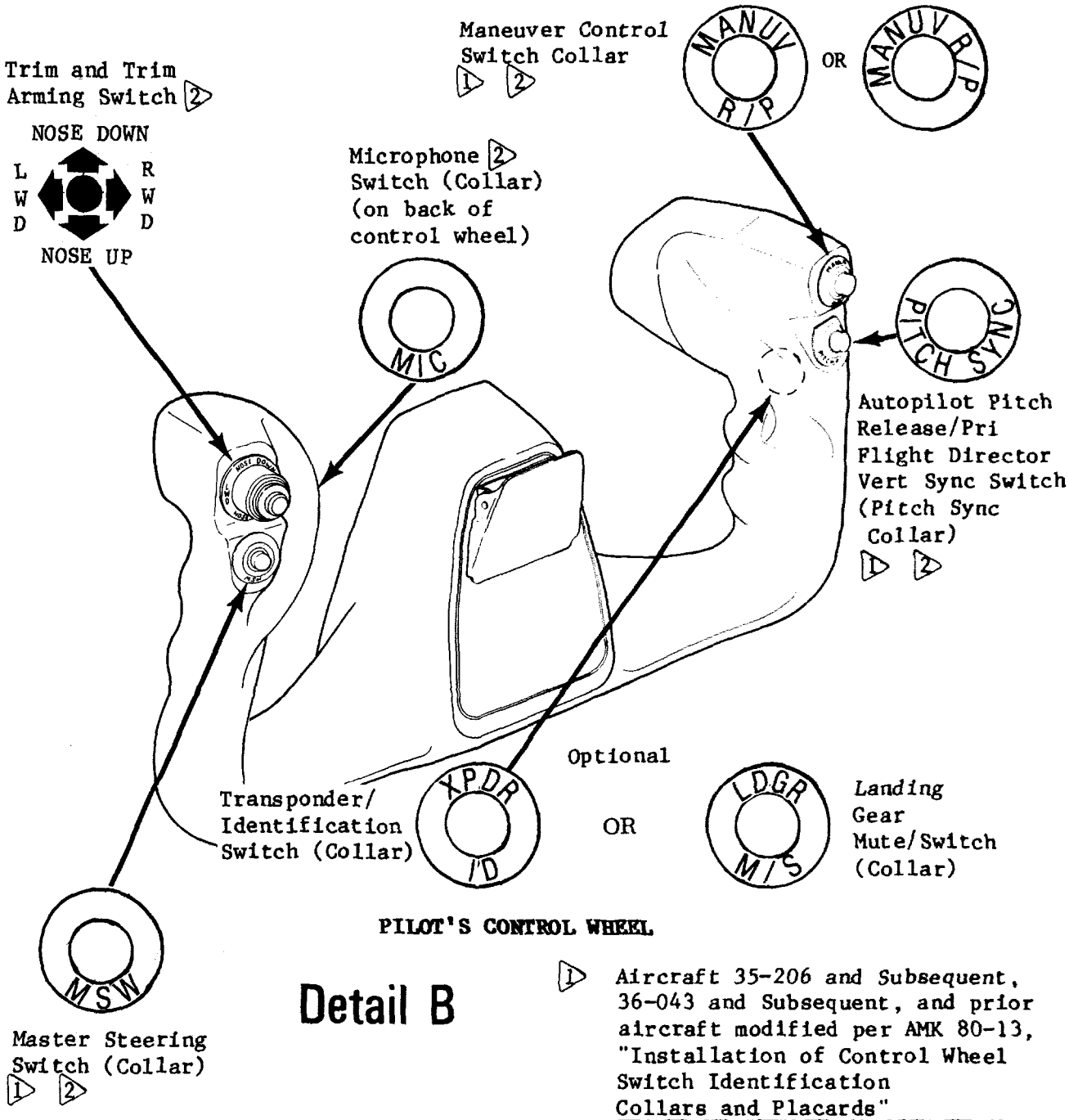
EFFECTIVITY: ALL

MM-99

* WHEN CONTOUR WHEEL IS INSTALLED, LOOP WIRING APPROXIMATELY 1-1/2 TURNS OUTSIDE CONTROL WHEEL HUB, THEN 1-1/2 TURNS INSIDE CONTROL WHEEL HUB. ENSURE THAT INDEX MARKS ALIGN.



Aileron Control System Rigging
Figure 201 (Sheet 2 of 5)



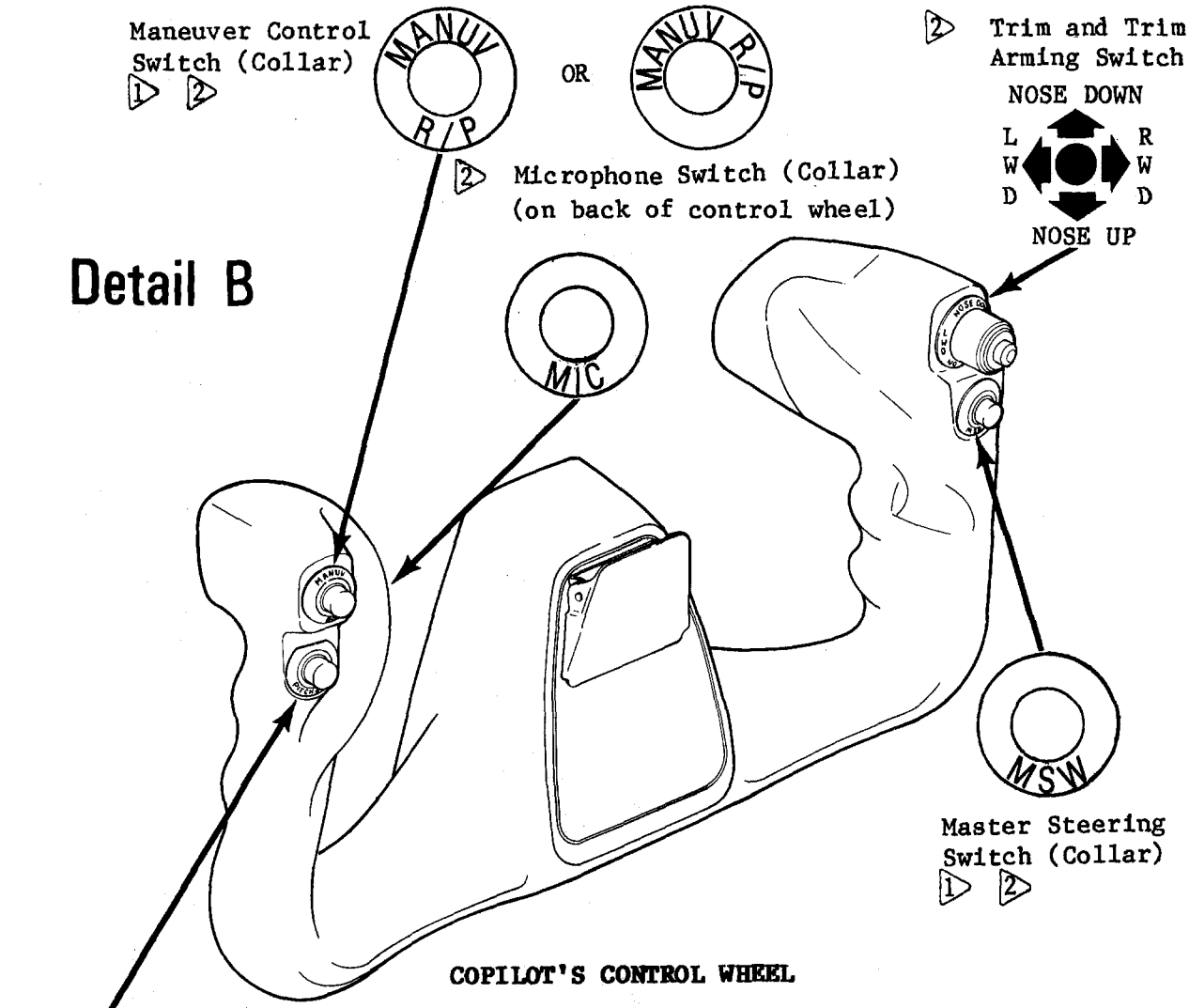
Detail B

1 Aircraft 35-206 and Subsequent, 36-043 and Subsequent, and prior aircraft modified per AMK 80-13, "Installation of Control Wheel Switch Identification Collars and Placards"

2 German Certified Aircraft

8-51B-3

Aileron Control System Rigging
Figure 201 (Sheet 3 of 5)



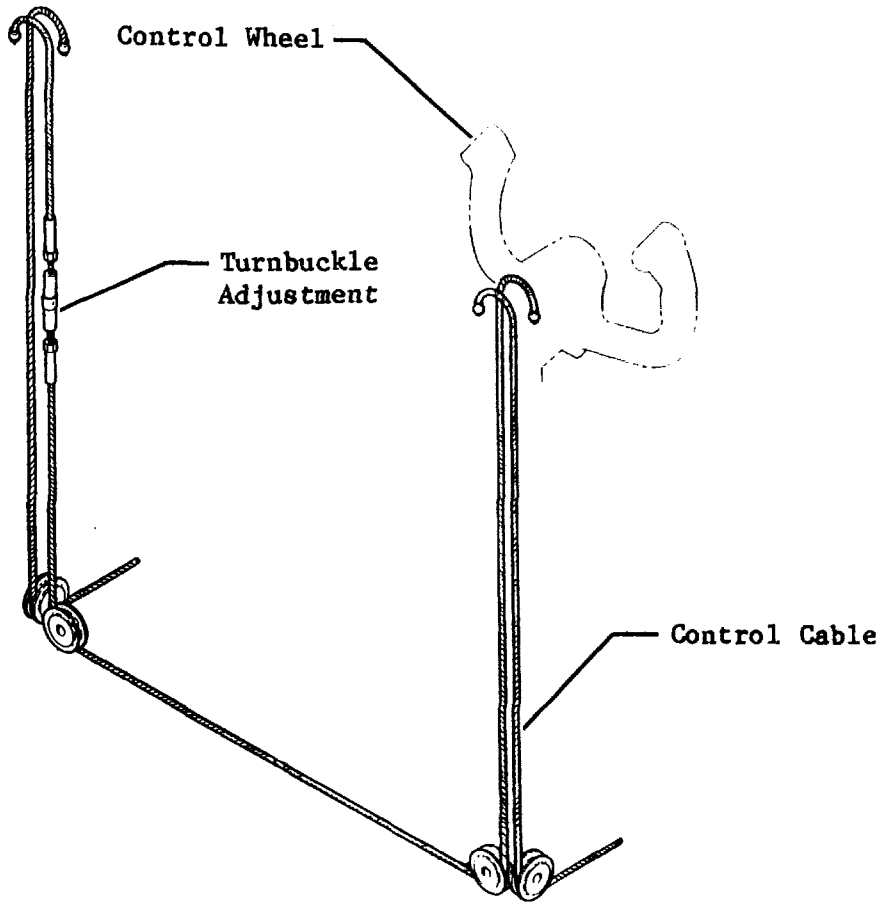
1 Aircraft 35-206 and Subsequent, 36-043 and Subsequent, and prior aircraft modified per AMK 80-13, "Installation of Control Wheel Switch Identification Collars and Placards

2 German Certified Aircraft

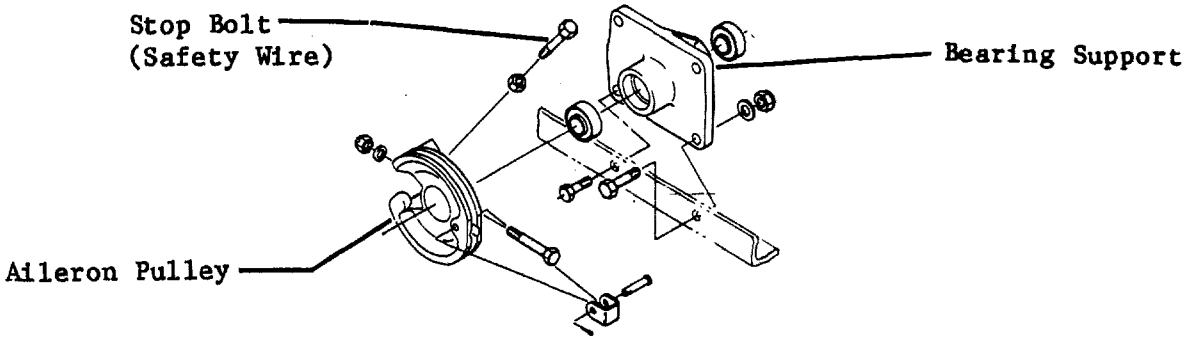
3 Aircraft 35-589 and Subsequent, 36-054 and Subsequent with dual flight directors

8-51B-3

Aileron Control System Rigging
Figure 201 (Sheet 4 of 5)

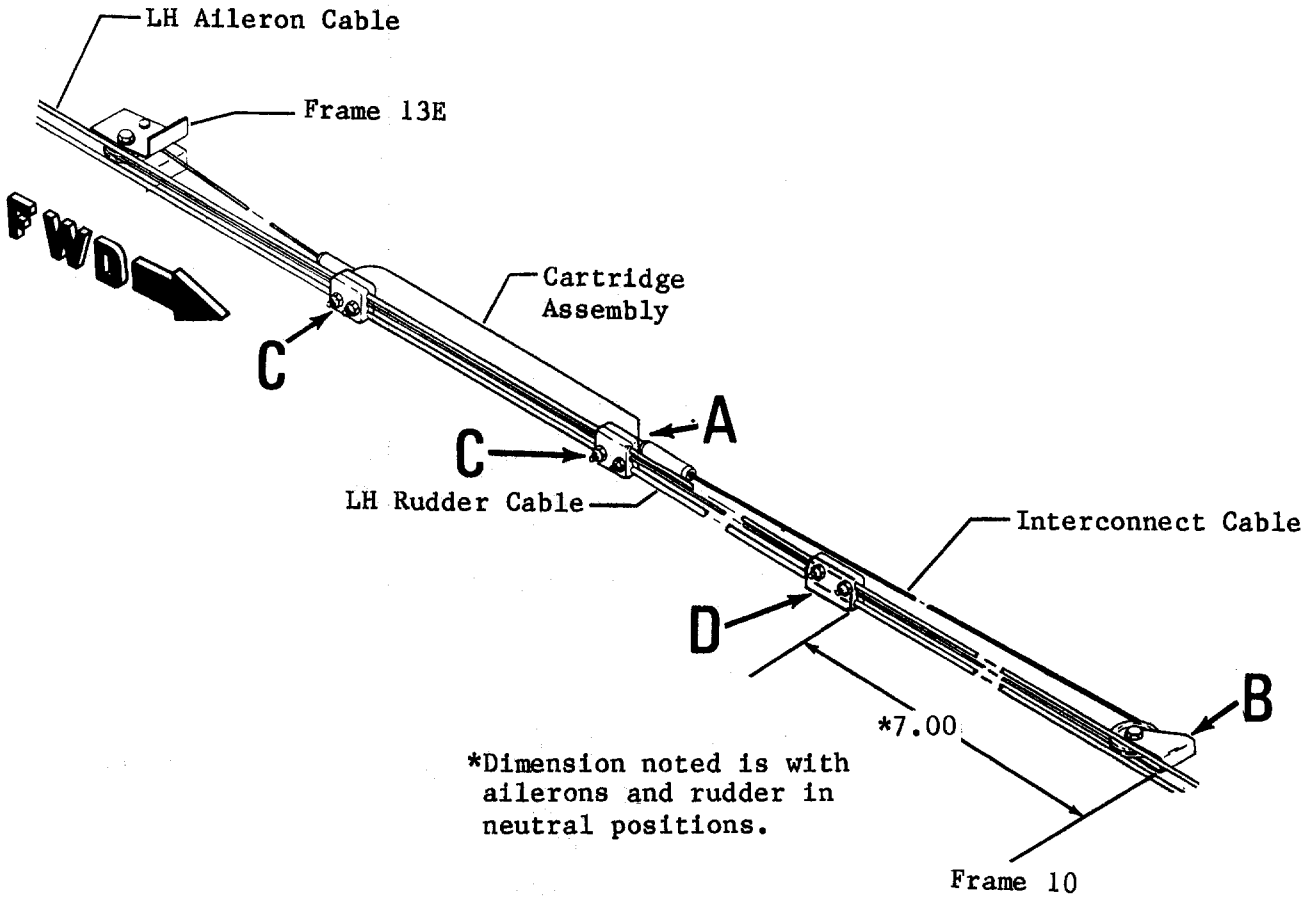


Control Column Cable Routing



Aileron Drive Sector

Aileron Control System Rigging
Figure 201 (Sheet 5 of 5)



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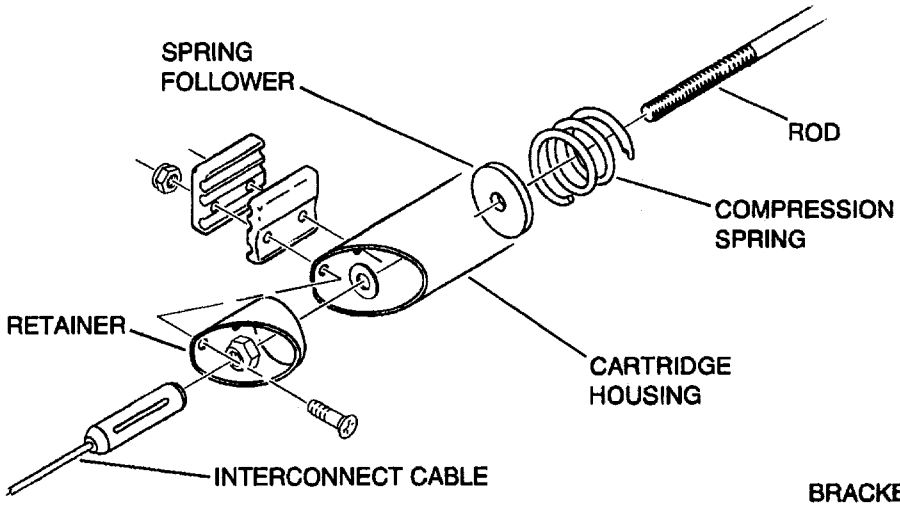
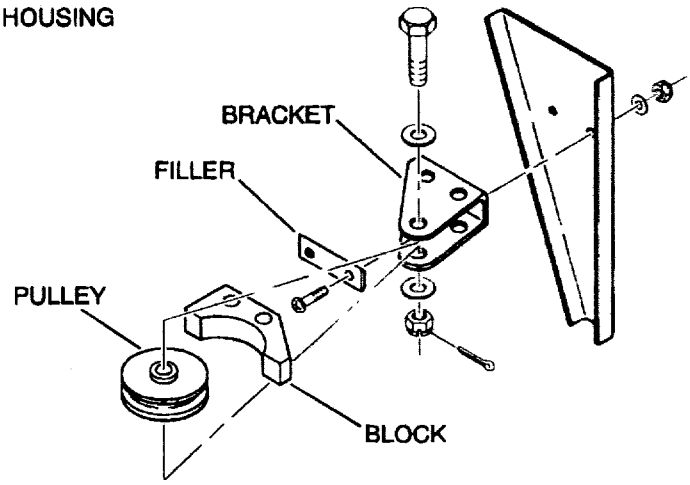
Aileron-Rudder Interrconnect
Figure 202 (Sheet 1 of 2)

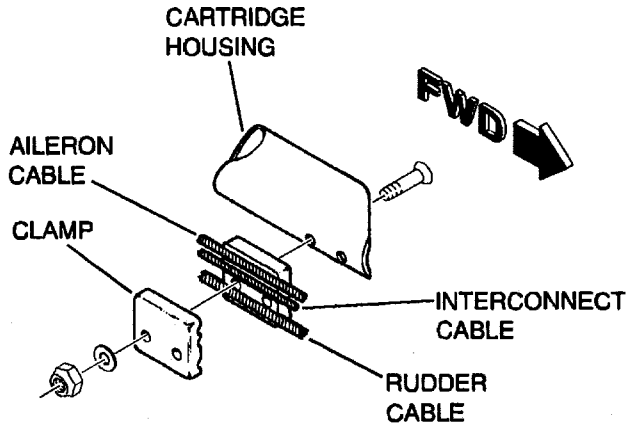


CARTRIDGE ASSEMBLY

Detail A



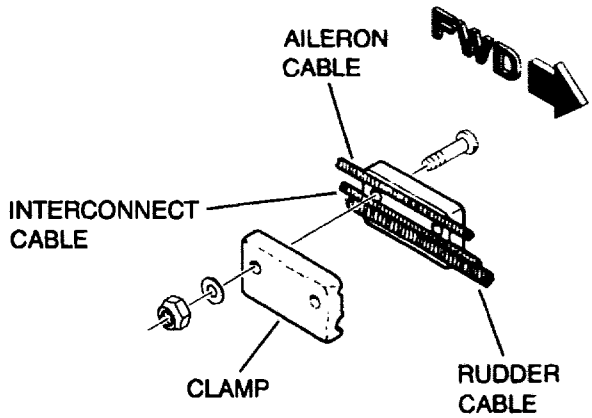
Detail B



ENSURE THAT AILERONS AND INTERCONNECT CABLES ARE ROUTED THROUGH CLEARANCE HOLES AND THAT RUDDER CABLE IS SECURE IN THIS CLAMP.

TYPICAL CABLE CLAMP

Detail C



ENSURE THAT RUDDER CABLE IS ROUTED THROUGH CLEARANCE HOLE AND THAT AILERON AND INTERCONNECT CABLES ARE SECURE IN THIS CLAMP.

Detail D

M35-271000-202-02

Aileron-Rudder Interconnect
Figure 202 (Sheet 2 of 2)

E. Adjust Aileron Control Cable Tension

NOTE: Prior to aileron control cable tension adjustment, torque wrench tensiometer adapter calibration must be accomplished. (Refer to Adjustment/Test, 27-00-00.)

Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The Cable Tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to Adjustment/Test, 27-00-00.)

(1) Remove forward lower fuselage access plate to gain access to LH and RH aileron forward control cable turnbuckles.
(2) With ailerons in faired position, insert rigging pin in aileron sector.
(3) Remove LH and RH wing access panels as required to gain access to LH and RH aileron wing control cable turnbuckles.
(4) Remove locking clips from turnbuckles.
(5) If a new cable is being installed, perform the following:
  (a) Adjust aileron cable tension to 200 (±10) pounds (90.7 [±4.5] kg).
  (b) Cycle aileron system 25 times to seat in and pre-stretch new cable.
  (c) Reduce cable tension.
(6) Adjust LH and RH aileron forward control cable tension to 75 (±10) pounds at 75°F (34.0 [±4.6] kg at 24°C).

NOTE: When autopilot roll servo cable tension is adjusted, primary aileron control cable aft of attach point will be 100 (±10) pounds (45.36 [±4.54] kilograms) and primary cable tension forward of attach point will be 75 (±10) pounds (34.02 [±4.54] kilograms).

(7) Adjust LH and RH aileron wing control cable tension to 75 (±10) pounds at 75°F (34.0 [±4.6] kg at 24°C).

NOTE: Wing aileron cable tension shall be measured at WS 45.00 through an available access panel in each wing at Spar 7 and aft.

Threads at opposite ends of turnbuckle barrel shall be started at the same time so that thread engagement on both ends will be approximately equal.

When cable turnbuckles are loosened or tightened to adjust cable tension, it may be necessary to readjust the ailerons to the droop neutral position. Repeat cable adjustments alternately until aileron is rigged to neutral at the required cable tension.

The final adjusted turnbuckle combination shall have no more than three threads of the terminal, fork, or eye exposed outside turnbuckle barrel.

(8) Once proper cable tension has been obtained, remove rigging pin and operationally check aileron control system. (Refer to Inspection/Check, this section.)
(9) Install locking clips in turnbuckles.
(10) Install all previously removed access covers, floorboards, and carpeting.
(11) Restore aircraft to normal.

4. Inspection/Check

A. Test Aileron Control System for Proper Operation

(1) Turn the control wheel to the right; the right aileron will move up and the left aileron will move down.
(2) Turn the control wheel back to neutral position; the ailerons will streamline.
(3) Turn the control wheel to the left; the left aileron will move up and the right aileron will move down.

B. Test Aileron Trim Tab for Proper Operation
(1) Depress pilot's Trim and Trim Arming Switch, and set to LWD (left wing down) position and hold.
(2) The aileron trim tab will move down from the aileron in a smooth, responsive movement.
(3) The aileron trim indicator on the center pedestal will indicate corresponding trim tab travel.
(4) Depress Pilot's Trim and Trim Arming Switch, and set to RWD (right wing down) position and hold.
(5) The aileron trim tab will move up from the aileron in a smooth, responsive movement.
(6) The aileron trim indicator will indicate corresponding trim tab travel.
(7) Repeat steps B.(1) thru B.(6) using copilot's Trim and Trim Arming Switch.

C. Perform Aileron Control Cable and Aileron-Rudder Interconnect Control Cable Tension Check

NOTE: Perform aileron control cable and aileron rudder-interconnect control cable tension check in accordance with the current inspection intervals specified in Chapter 5.

Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The cable tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to 27-00-00, Adjustment/Test.)

(1) Position ailerons to the neutral position.
(2) Remove floorboards as required to gain access to aileron forward control cables.
(3) Attach tensiometer to LH forward aileron control cable. Cable tension's shall be 75 (±10) pounds at 75°F (34.0 [±4.6] kg at 24°C). Check RH forward aileron control cable in the same manner.

NOTE: When autopilot roll servo cable tension is adjusted, primary aileron control cable aft of attach point will be 100 (±10) pounds (45.36 [±4.54] kilograms) and primary cable tension forward of attach point will be 75 (±10) pounds (34.02 [±4.54] kilograms).

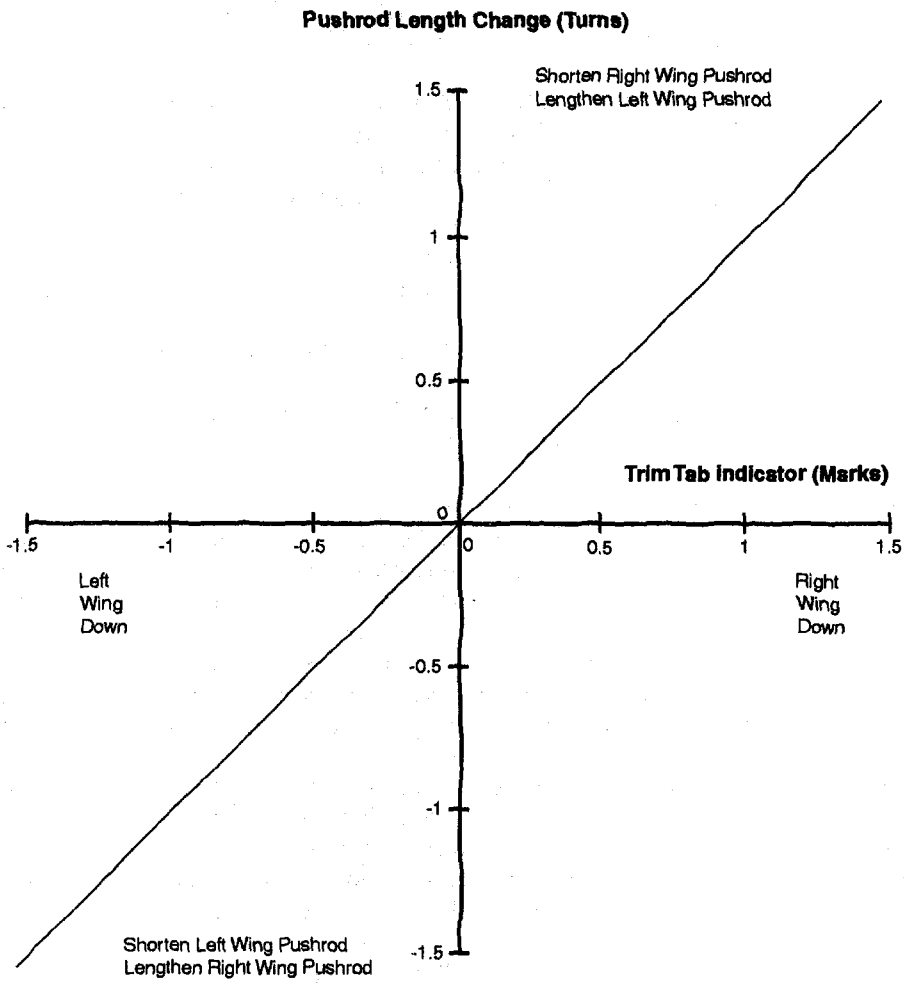
(4) With both the ailerons and the rudder in neutral, attach tensiometer to aileron-rudder interconnect control cable. Cable tension's shall be 25 (±5) pounds (11.34 [±2.27] kilograms) at 75°F (24°C).
(5) Remove LH and RH wing access panels as required to gain access to aileron wing control cables.
(6) Attach tensiometer to LH aileron wing control cable. Cable tension's shall be 75 (±10) pounds at 75°F (34.0 [±4.6] kg at 24°C). Check RH aileron wing control cable in the same manner.

NOTE: Wing aileron cable tension shall be measured at WS 45.00 through an available access panel in each wing at Spar 7 and aft.

The aileron cable must be lying against the back of the tensiometer when pressure is applied so that no extra pull is applied to the cable.

Avoid pushing or pulling on tensiometer or cable while a reading is being taken.

(7) Adjust aileron control and aileron-rudder interconnect cables if readings do not fall within tolerances. (Refer to Adjustment/Check, this section.)
(8) Remove tensiometer from aileron control cable.
(9) Install all floorboards and wing access covers.
(10) Restore aircraft to normal.



Aileron Balance Tab Rigging
Figure 203

AILERON - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Torque Wrench		Commercially available	Torquing.
Safety Wire	MS20995NC32	Commercially available	Safetying.
Rivets	AD32ABS	Commercially available	Installing aileron.
Grease	No. 33	Dow Corning Midland, MI	Installing aileron.

2. Removal/Installation

NOTE: The following procedures require that the hinge bolt access doors and the aileron control wing gap door to be removed.

Removal and installation procedures for the left and right ailerons differ in that the trim tab actuator and potentiometer wires are installed in the left aileron.

When installing a new aileron, it is necessary to trim both ends of the aileron as well as install components from the old aileron on the new aileron. Two aileron installation procedures have been provided.

A. Removal of Aileron (See Figure 201.)
   (1) Disconnect external electrical power source from aircraft.
   (2) Remove hinge bolt access door from underside of aileron and aileron control wing gap door from wing.
   (3) Disconnect aileron trim tab actuator/potentiometer electrical wires at electrical wiring splice inside wing and pull electrical wiring out through aft wing spar.
   (4) Disconnect balance tab push-pull tubes from wing bracket.
   (5) Remove nut, washers, and clevis bolt connecting drive clevis to yoke. As bolt is removed, shims will be released. Save shims for reinstallation.

CAUTION: AILERON MUST BE SUPPORTED WHEN HINGE BOLTS ARE REMOVED OR INSTALLED. SIDE LOADS ON HINGE BEARINGS DURING REMOVAL AND INSTALLATION CAN DAMAGE HINGE BEARINGS.

   (6) Remove hinge bolts securing aileron to wing.
   (7) Disconnect aileron bonding jumper at wing and remove aileron.
   (8) If aileron is to be replaced, remove yoke bolt, yoke, and shims from center hinge casting.

B. Installation of Existing Aileron (See Figure 201.)

CAUTION: ENSURE THAT LOWER BEND RELIEF IN INBOARD AILERON RIB IS NOT OBSTRUCTED OR SEALED CLOSED. WATER ACCUMULATION IN AILERON MAY FREEZE AND AFFECT AILERON BALANCE.

(1) If any portion of a control surface has been repainted, repaired or replaced, a check must be made to determine if the balance is still within the prescribed limits. (Refer to 27-00-00.)

NOTE: Ensure that aileron hinge bearings rotate smoothly and are correctly installed before installing aileron.

WARNING: MAKE SURE THE BONDING JUMPER DOES NOT INTERFERE WITH AILERON DRIVE YOKE WHEN INSTALLED. AILERON TRAVEL WILL BE LIMITED IF BONDING JUMPER HOOKS ON THE DRIVE YOKE.

IF BONDING JUMPER IS REPLACED, MAKE SURE IT IS REPLACED WITH THE SAME TYPE AND LENGTH. (REFER TO APPLICABLE IPC.)

CAUTION: AILERON MUST BE SUPPORTED WHEN HINGE BOLTS ARE REMOVED OR INSTALLED. SIDE LOADS ON HINGE BEARINGS DURING REMOVAL AND INSTALLATION CAN DAMAGE HINGE BEARINGS.

USE EXTREME CARE TO AVOID OVERTORQUING HINGE BOLT OR EXCESSIVE CONTROL FORCE MAY BE REQUIRED.

(2) Align aileron on wing and connect aileron bonding jumper to aileron hinge fitting in wing. Secure aileron to wing with hinge bolts and torque bolts to 30 to 40 inch-pounds [3.4 to 4.5 Nm]. Safety wire bolts.
(3) Align aileron drive yoke with aileron drive clevis. Shim as necessary between yoke and drive clevis to prevent bearing preloading when clevis bolt is installed. Install clevis bolt and nut. Torque nut 30 to 40 inch-pounds [3.4 to 4.5 Nm] and install cotter pin.
(4) Route aileron trim tab actuator and potentiometer electrical wiring through wing rear spar and splice at bundle.
(5) Connect balance tab push-pull tubes to wing bracket.
(6) Check aileron for proper operation.
(7) Check aileron rigging. (Refer to 27-10-00, Adjustment/Test.)
(8) Install all remaining access doors.
(9) Touch up paint.
(10) Connect external electrical power source to aircraft.

C. Installation of New Aileron (See Figure 201.)

CAUTION: ENSURE THAT LOWER BEND RELIEF IN INBOARD AILERON RIB IS NOT OBSTRUCTED OR SEALED CLOSED. WATER ACCUMULATION IN AILERON MAY FREEZE AND AFFECT AILERON BALANCE.

(1) Install yoke. Shim yoke top and bottom as necessary to ensure there is no end play of yoke in aileron and secure with yoke bolt, nut, and washer. Torque yoke bolt to 30 to 40 inch-pounds [3.4 to 4.5 Nm]. Safety wire bolt shank to center hinge casting.

WARNING: MAKE SURE THE BONDING JUMPER DOES NOT INTERFERE WITH AILERON DRIVE YOKE WHEN INSTALLED. AILERON TRAVEL WILL BE LIMITED IF BONDING JUMPER HOOKS ON THE DRIVE YOKE.

IF BONDING JUMPER IS REPLACED, MAKE SURE IT IS REPLACED WITH THE SAME TYPE AND LENGTH. (REFER TO APPLICABLE IPC.)

CAUTION: AILERON MUST BE SUPPORTED WHEN HINGE BOLTS ARE REMOVED OR INSTALLED. SIDE LOADS ON HINGE BEARINGS DURING REMOVAL AND INSTALLATION CAN DAMAGE HINGE BEARINGS.

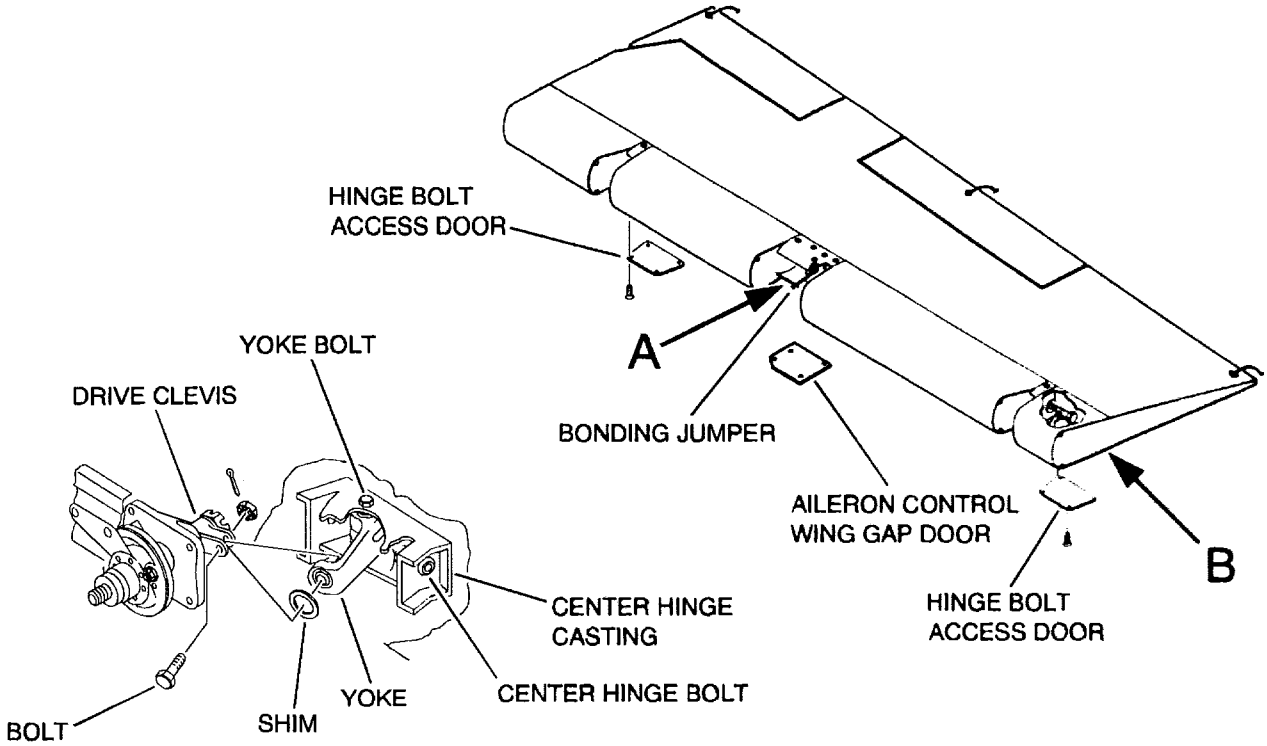
***USE EXTREME CARE TO AVOID OVERTORQUING HINGE BOLT OR EXCESSIVE* CONTROL FORCE MAY BE REQUIRED.**

(2) Align aileron on wing hinge brackets and insert inboard, center, and outboard hinge bolts.
(3) Lay out trim lines on inboard and outboard ends of aileron as necessary to attain proper clearances as shown.
(4) Check all hinge points for proper alignment and freedom of movement.
(5) Remove aileron from wing and trim excess material from inboard and outboard ends as determined in step (3).
(6) Brush seals are temporarily installed. Install remaining rivets (P/N AD32ABS) securing brush seal to outboard rib.

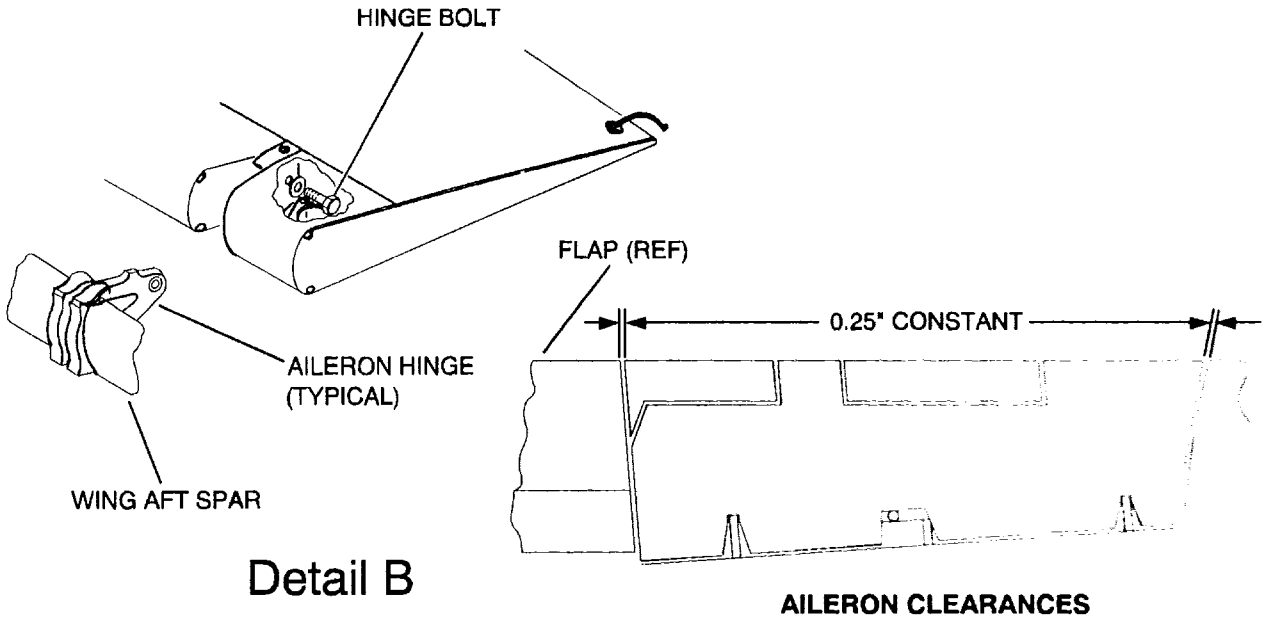
NOTE: *On Aircraft incorporating SSK 938*, the aileron brush seal is attached to the wing extension and requires no further assembly.

(7) Install aileron hinge bolts and hinge bolt doors.
(8) Mask off hinge and drive bearings. Paint aileron to match aircraft colors.
(9) Check aileron balance. (Refer to 27-00-00.)
(10) Apply a thin coat of Dow Corning No. 33 grease to aileron brush seal.
(11) Remove hinge bolt access doors and align aileron on wing. Connect aileron bonding jumper to aileron hinge fitting in wing. Secure aileron to wing with hinge bolts and torque bolts to 30 to 40 inch-pounds [3.4 to 4.5 Nm]. Safety wire bolts.

(12) Align yoke with drive clevis. Shim as necessary between yoke and drive clevis to prevent bearing preloading when clevis bolt is installed. Install clevis bolt and nut. Torque nut 30 to 40 inch-pounds [3.4 to 4.5 Nm] and install cotter pin.
(13) Route aileron trim tab actuator and potentiometer electrical wiring through the wing rear spar and splice at bundle.
(14) Connect balance tab push-pull tubes to wing bracket.
(15) Check aileron for proper operation.
(16) Check aileron rigging. (Refer to 27-10-00 Adjustment/Test.)
(17) Install all remaining access doors.
(18) Connect external electrical power source to aircraft.



Detail A



Detail B

AILERON CLEARANCES

M35-271001-201-01

Aileron Installation
Figure 201

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

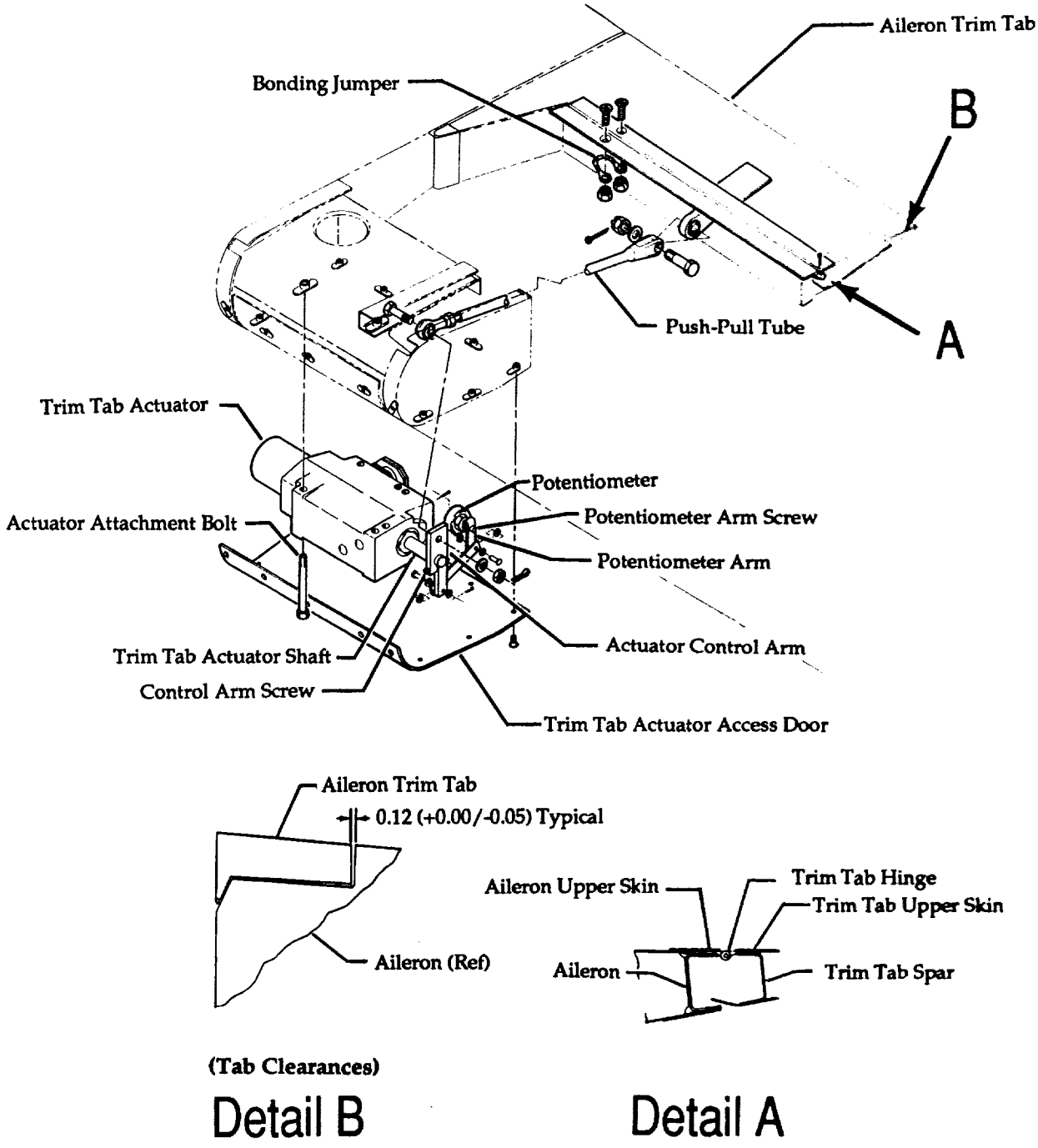
NAME	PART NUMBER	MANUFACTURER	USE
Fastener	As Required	CLECO	Hold Trim Tab

2. Removal/Installation

NOTE:
- The aileron trim tab is attached to the left aileron by a continuous-type hinge. It is powered by a rotary-type electrical actuator, located in the aileron leading-edge.
- After a new trim tab has been installed, the aileron must be rebalanced. (Refer to 27-00-00.)
- Access to the trim tab actuator is through the trim tab actuator access door on the aileron leading-edge.

A. Remove Aileron Trim Tab. (See Figure 201.)
   (1) Remove aileron from wing. (Refer to 27-10-01.)
   (2) Disconnect trim tab actuator push-pull tube and bonding jumper from trim tab.
   (3) Using a No. 30 (0.128 inch, 3.25 millimeter) bit, drill out rivets securing trim tab hinge to aileron spar. Remove trim tab from aileron.

B. Install Aileron Trim Tab. (See Figure 201.)
   (1) Install new trim tab on aileron with trim tab hinge between aileron rear spar flange and skin. Adjust trim tab for proper clearance between trim tab and aileron, as shown.
   (2) Place a straightedge on aileron trailing edge. Check to ensure that trim tab trailing-edge and aileron trailing-edge are aligned.
   (3) With trim tab properly positioned, mark rivet locations on outboard and inboard ends of trim tab hinge. Remove trim tab and drill a No. 30 hole at each location mark.
   (4) Install trim tab and cleco into place. Using remaining tab hinge holes in aileron as a pattern, drill a No. 30 hole at each location except for the fourth hole from the outboard end of the hinge. Drill a No. 19 (0.166 inch, 4.22 millimeter) hole at this location. This is for bonding jumper attachment.
   (5) Secure trim tab to aileron with rivets (P/N MS20426AD4-6).
   (6) Connect trim tab bonding jumper and push-pull tube to trim tab.
      (a) Check electrical resistance between trim tab and aircraft structure. Resistance shall NOT be greater than the value specified in Chapter 20 of the Wiring Manual.
   (7) Check aileron balance. (Refer to 27-00-00.)
   (8) Install aileron. (Refer to 27-10-01.)



4-4C-2

Aileron Trim Tab Installation
Figure 201

3. Inspection/Check

A. Tools and Equipment

NOTE:
- Dial indicator must be calibrated within 12 months after being placed into service and every 12 months therafter.
- Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Aileron/Rudder Trim Tab Free Play Tool	2370120-1	Learjet Corp. Wichita, KS	Hold Dial Indicator
Aileron/Rudder Trim Tab Free Play Tool	2370120-2	Learjet Corp. Wichita, KS	Hold torque wrench
Torque wrench Click-type	0 - 100 in/lbs.	Commercially Available	Apply pressure to trim tab

B. Measure Aileron Trim Tab Free Play. (See Figure 202.)

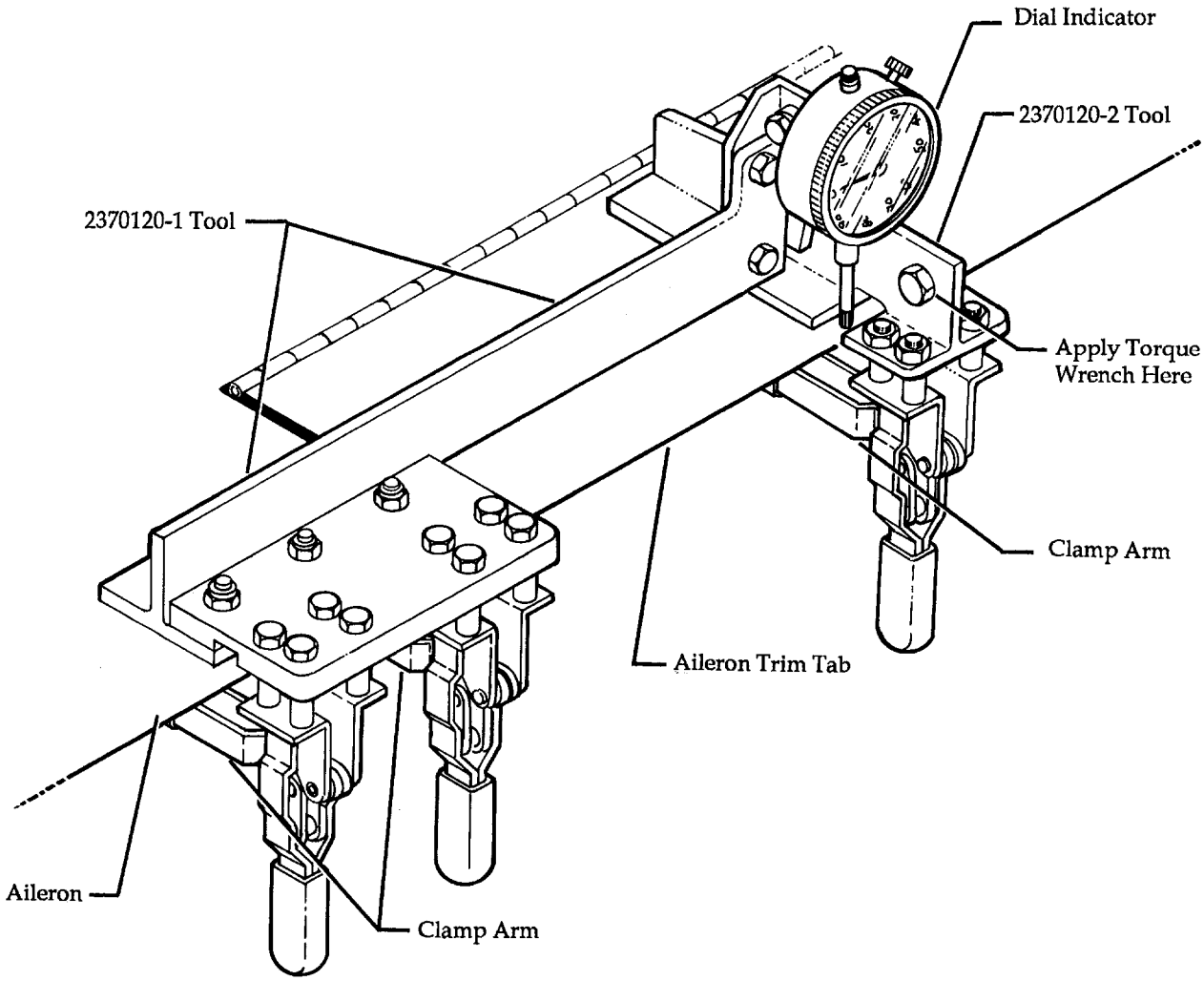
NOTE: Perform aileron trim tab free play measurement in accordance with the current inspection intervals specified in Chapter 5.

(1) Block left aileron to prevent movement.
(2) Streamline aileron trim tab.
(3) Position -1 Aileron/Rudder Trim Tab Free Play Tool on aileron.
  (a) Adjust pads on clamp arms to apply adequate pressure on aileron to hold tool in place.
(4) Position -2 Aileron/Rudder Trim Tab Free Play Tool on aileron trim tab as close as possible to dial indicator. (See Figure 202.)
  (a) Adjust pads on clamp arm to apply sufficient clamping force to hold tool in place.

NOTE:
- Aileron/Rudder Trim Tab Free Play Tools shall not rock.
- Do not apply enough force to permanently deform aileron or trim tab surface.

(5) Zero dial indicator.
(6) With slight finger pressure only, check for excessive free play.
  (a) If free play is more than 0.020 inch in either direction, check trim tab actuator linkage for proper adjustment.
  (b) Adjust or replace aileron trim tab actuator linkage components to eliminate free play. (Refer to 27-10-06, Removal/Installation.)
(7) Set torque wrench to 35 inch pounds and lock.
(8) Position torque wrench on 7/16 bolt head on top of -2 Aileron/Rudder Trim Tab Free Play Tool with handle perpendicular to control surface. (See Figure 202.)
(9) Verify that dial indicator is at zero and slowly apply upward pressure to wrench. Watching dial indicator, note and record reading when torque wrench releases. (First reading.)
(10) Repeat step (9) three times to verify results, recording each reading. (See Figure 203.)

(11) Average closest two readings. (Example: 0.016, 0.019 and 0.017; eliminate 0.019 and add 0.016 and 0.017 and divide by 2.) (See Figure 203.)
(12) Repeat steps (7), (8), (9), (10) and (11) applying downward pressure to torque wrench.
(13) Add measured free play in both directions to determine total. If total free play exceeds 0.050 inch, replace trim tab hinge pin and repeat inspection. If hinge pin has been replaced, replace trim tab hinge and repeat inspection.
(14) After free play measurement is complete remove Aileron/Rudder Trim Tab Free Play Tools from aircraft and unblock aileron.
(15) Return aircraft to normal.



14-232B

Aileron Trim Tab Free Play Measurement
Figure 202

Learjet

AILERON TRIM TAB FREE PLAY WORKSHEET

MEASURE UP FREE PLAY

DIAL INDICATOR READING NO. 1 _____

DIAL INDICATOR READING NO. 2 _____

DIAL INDICATOR READING NO. 3 _____

ADD CLOSEST TWO READINGS _____ + _____ = _____

DIVIDE TOTAL BY 2 _____ / 2 = _____ (UP FREE PLAY)

MEASURE DOWN FREE PLAY

DIAL INDICATOR READING NO. 1 _____

DIAL INDICATOR READING NO. 2 _____

DIAL INDICATOR READING NO. 3 _____

ADD CLOSEST TWO READINGS _____ + _____ = _____

DIVIDE BY 2 _____ / 2 = _____ (DOWN FREE PLAY)

TOTAL FREE PLAY

UP FREE PLAY _____

DOWN FREE PLAY + _____

TOTAL FREE PLAY = _____

NOTE: Total Aileron Trim Tab free play shall not exceed 0.050 inch.

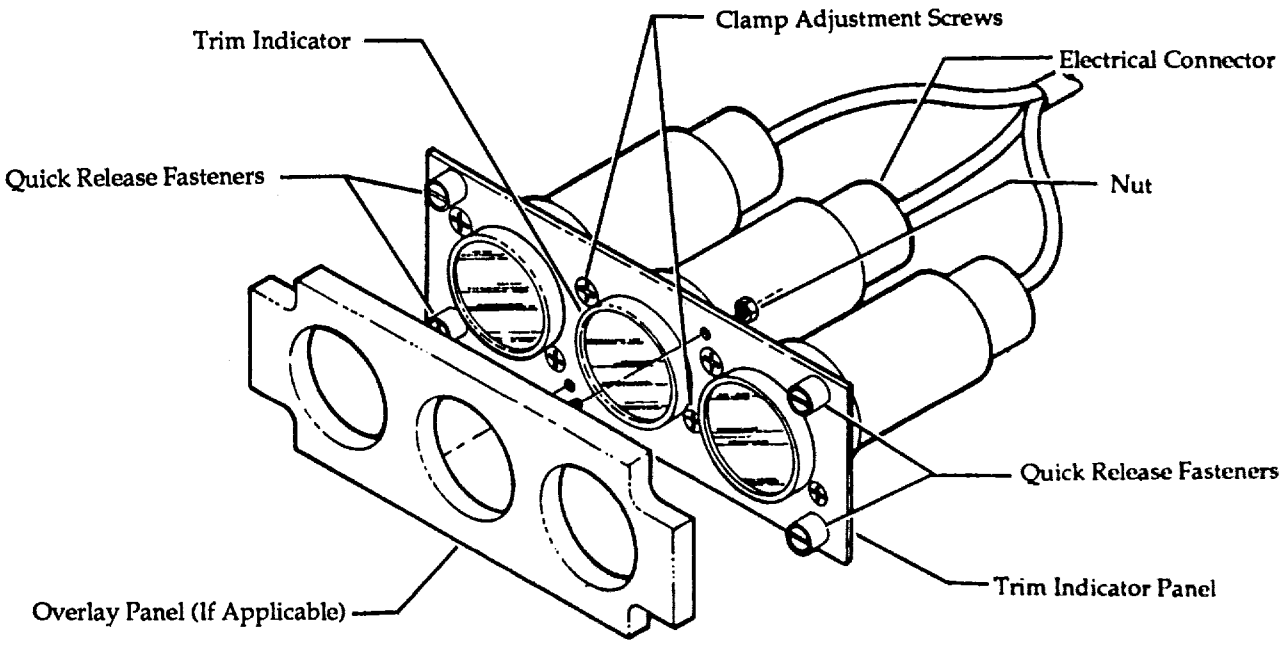
Aileron Trim Tab Free Play Worksheet
Figure 203

1. **Removal/Installation** (See figure 201.)
   A. Remove Aileron Trim Indicator
      (1) Remove electrical power from aircraft.
      (2) Release trim panel assembly from the pedestal assembly by loosening quick release fasteners.
      (3) If applicable, remove parts securing overlay panel to face of trim panel and remove overlay.
      (4) Disconnect electrical connector from aileron trim indicator.
      (5) Loosen clamp adjustment screw and remove aileron trim indicator from trim indicator panel.
   B. Install Aileron Trim Indicator
      (1) Position aileron trim indicator in trim indicator panel and secure by tightening clamp adjustments screw.
      (2) Connect electrical connector to aileron trim indicator.
      (3) If applicable, position overlay panel on face of trim indicator panel and secure with attaching parts.
      (4) Position trim indicator panel in center pedestal and secure with quick release fasteners.
      (5) Restore electrical power to aircraft.
      (6) Perform Adjustment/Test of aileron trim indicator.

2. **Adjustment/Test** (See figure 202.)

   NOTE:
   - The aileron trim position indicator, located on the cockpit center pedestal, indicates aileron trim tab travel. Indicator input is from a potentiometer attached to the aileron trim tab actuator.
   - Calibration of the aileron trim indicator is performed at the position indicator calibration assembly located inside of the trim switch panel.
   - The aileron trim tab potentiometer must be approximately centered with tab in neutral position prior to calibrating aileron trim indicator.

   A. Calibrate aileron trim position indicator as follows:
      (1) Loosen trim switch panel fasteners and raise switch panel sufficiently to gain access to calibration assembly.
      (2) Actuate aileron trim tab to neutral position.
      (3) Adjust Roll Zero potentiometer to obtain zero indication on roll trim indicator.
      (4) Actuate aileron trim tab to full up position.
      (5) Adjust Roll Expand potentiometer to obtain a right wing down indication on the aileron trim indicator.
      (6) Actuate aileron trim tab to full down position.
      (7) Adjust Roll Expand potentiometer to obtain left wing down indication on the aileron trim indicator.
      (8) Cycle aileron trim tab while checking respective indications on roll trim indicator. Readjust Roll Expand potentiometer as necessary to obtain coordinated indications.
      (9) Position trim switch panel in pedestal and secure fasteners.



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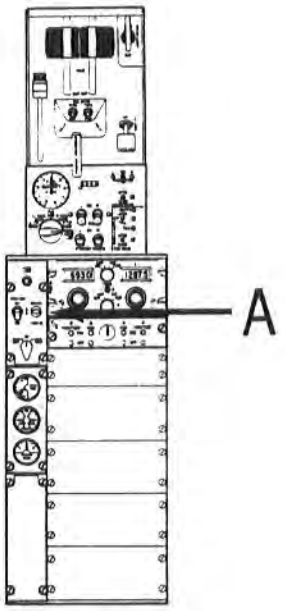
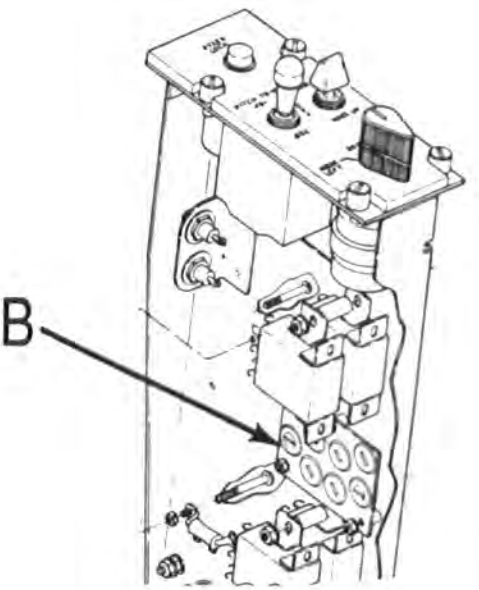
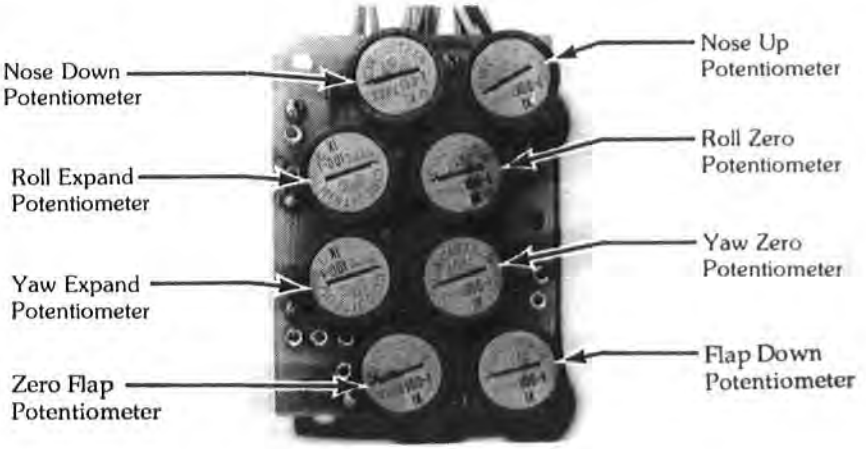
Aileron Trim Indicator Installation
Figure 201



(TYPICAL)



Detail A



Detail B

Position Indicator Calibration Assembly
Figure 202

AILERON BALANCE TAB - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Aileron Balance Tab (See Figure 201.)

(1) Remove aileron from wing. (Refer to Removal/Installation, 27-10-01.)

(2) Disconnect aileron balance tab push-pull tubes and bonding jumper from balance tab.

(3) Using a No. 30 drill, drill out rivets securing balance tab hinge to aileron spar. Remove balance tab from aileron.

B. Install Aileron Balance Tab (See Figure 201.)

(1) When installing a new aileron balance tab, or a balance tab which has had the hinge replaced, and not been drilled, proceed as follows:

(a) Position aileron balance tab on aileron with balance tab hinge between aileron rear spar flange and skin. Adjust balance tab for proper clearance between balance tab and aileron as shown. Place a straightedge on aileron trailing edge and check to ensure that balance tab trailing edge and aileron trailing edge are aligned. Secure balance tab in place with adhesive tape.

(b) With balance tab properly positioned, mark rivet locations through aileron skin onto outboard and inboard ends of balance tab hinge. Remove balance tab and drill a No. 30 hole, through hinge half, at each location mark.

(c) Position balance tab on aileron trailing edge, aligning drilled holes in hinge with existing holes in aileron, and secure in place with Cleco's. Using remaining tab hinge holes in aileron as a pattern, drill a No. 30 hole at each location.

(d) Secure balance tab to aileron with rivets (P/N MS20426AD4-6).

(2) When installing a previously removed aileron balance tab, which has hinge half drilled, proceed as follows:

(a) Position balance tab on aileron trailing edge, aligning drilled holes in hinge with holes in aileron, and secure in place with Cleco's.

(b) Place a straightedge on aileron trailing edge and check to ensure that balance tab trailing edge and aileron trailing edge are aligned.

NOTE: If aileron and aileron balance tab trailing edges do not align, aileron balance tab hinge shall be replaced, realigned to aileron, and redrilled in accordance with previous step B.(1).

(c) Secure balance tab to aileron with rivets (P/N MS20426AD4-6).

(3) Check aileron balance tab for freedom of movement.

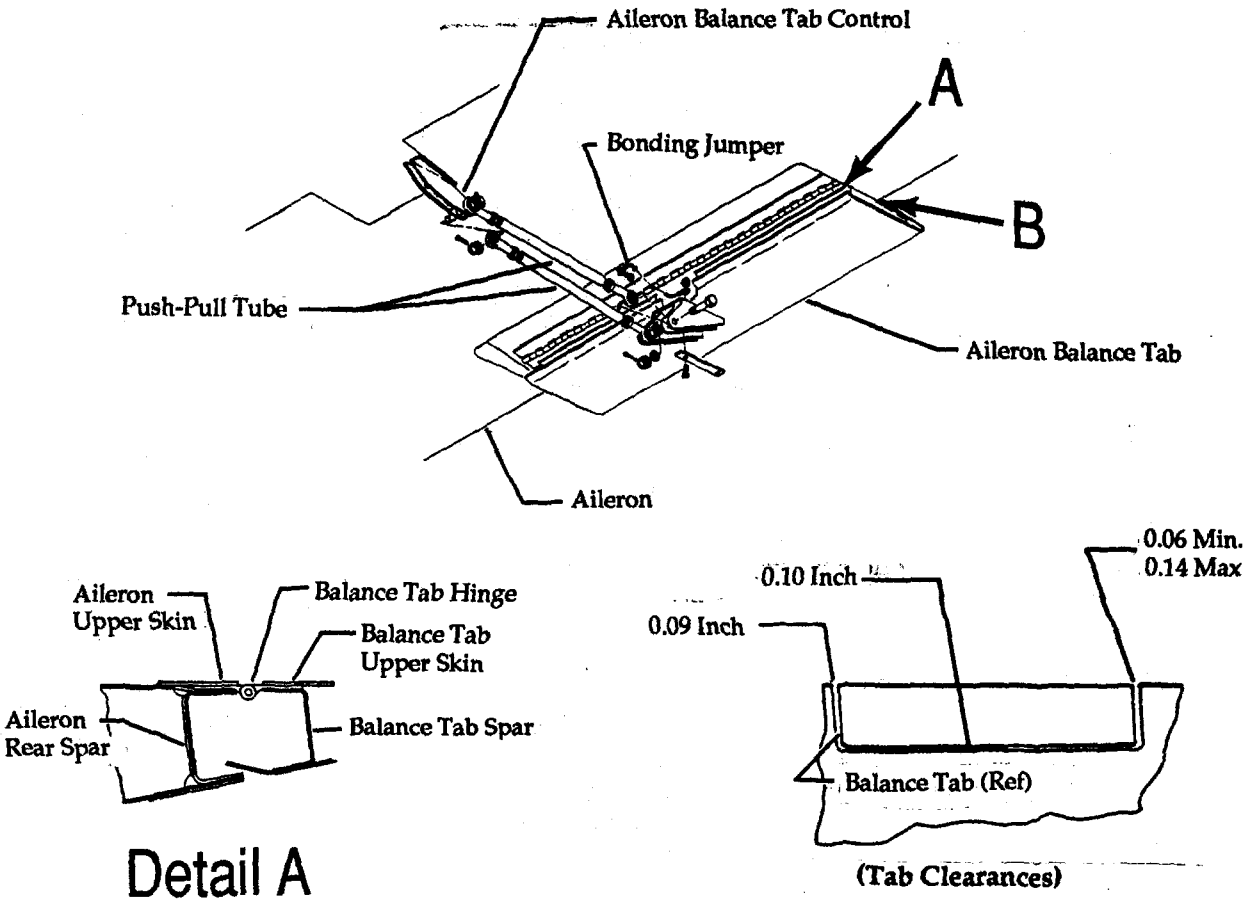
(4) Connect balance tab bonding jumper and push-pull tubes to balance tab. Check electrical bonding. (Refer to Wiring Manual, Chapter 20.)

(5) Check aileron balance. (Refer to Control Surface Balancing, 27-00-00.)

(6) Install aileron. (Refer to Removal/Installation, 27-10-01.)

(7) Rig aileron balance tab. (Refer to Adjustment/Test, 27-10-00.)

(8) Restore aircraft to normal.



Aileron Balance Tab Installation
Figure 201

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Methyl-Ethyl-Ketone (MEK)	Spec. TT-M-261	Commercially Available	Surface prep
Clean, lint-free cotton cloth		Commercially Available	Cleaning
1,1,1 trichloroethane	Spec. MIL-T-81533	Commercially Available	Surface prep
Adhesive (Retaining Compound)	RC680	Loctite Corp. Newington, CT	Control wheel and bearing installation
Adhesive	RTV-108	General Electric Co. Waterford, NY	Washer installation
Primer	1200	Dow Corning Co. Midland, MI	Washer installation

2. Removal/Installation

NOTE: Inspect control wheel assemblies in accordance with current inspection intervals specified in Chapter 5.

A. Remove Control Wheel Assembly. (*Aircraft 35-002 thru 35-487 and 36-002 thru 36-050*) (See Figures 201 and 203.)

NOTE: Disassemble control wheel assembly only to extent necessary to complete needed repairs.

(1) Remove control wheel sector cover and control wheel escutcheon.
(2) Remove nut and washer attaching control wheel to control wheel shaft and remove control wheel from control column.
(3) If control wheel is not to be removed further, support wheel to prevent damage to electrical wiring and proceed to step (4). If control wheel is to be replaced, proceed as follows:
  (a) Disconnect plug P80 located forward of control column base.
  (b) Identify the pins of plug P80 which connect to control wheel to be replaced. (Refer to Wiring Manual.)
  (c) Remove applicable pins from plug P80.
  (d) Pull control wheel away from control column, pulling electrical wiring through column.
(4) Remove nut and washer attaching control wheel shaft to control column assembly.
(5) Remove adhesive from around phenolic washer on aft side of control column assembly and remove washer.
(6) Remove screws attaching aft bearing plate to control column and remove bearing plate.
(7) Remove floorboards to gain access to aileron sector. Remove forward aileron control cables from aileron sector.
(8) On *Aircraft 35-002 thru 35-017 and 36-002 thru 36-008*, remove control wheel sector spring pin.

(9) Remove shaft and aft bearing from control column.
(10) On *Aircraft 35-018 thru 35-487 and 36-009 thru 36-050*, remove control wheel washers and spacers.
(11) Remove spring pins that hold control cables in place.
(12) Remove control cables. Tag each cables position for proper reinstallation.
(13) Remove control wheel sector.
(14) Remove screws and nuts attaching forward bearing plate. Note position of screws. The larger screw is in the aileron stop and must be reinstalled in the proper position.
(15) Remove bearing plate.
(16) Press out forward bearing.

B. Install Control Wheel Assembly. *(Aircraft 35-002 thru 35-487 and 36-002 thru 36-050)* (See Figures 201 and 203.)
(1) Clean aft face of control column fork, aft bearing plate and attaching screws with MEK. Wipe dry with a clean dry cloth. Do not allow MEK to air dry as a residue remains.

CAUTION: WHEN INSTALLING AFT BEARING, ENSURE THAT RETAINING COMPOUND IS NOT APPLIED TO INNER SURFACE OF BEARING.

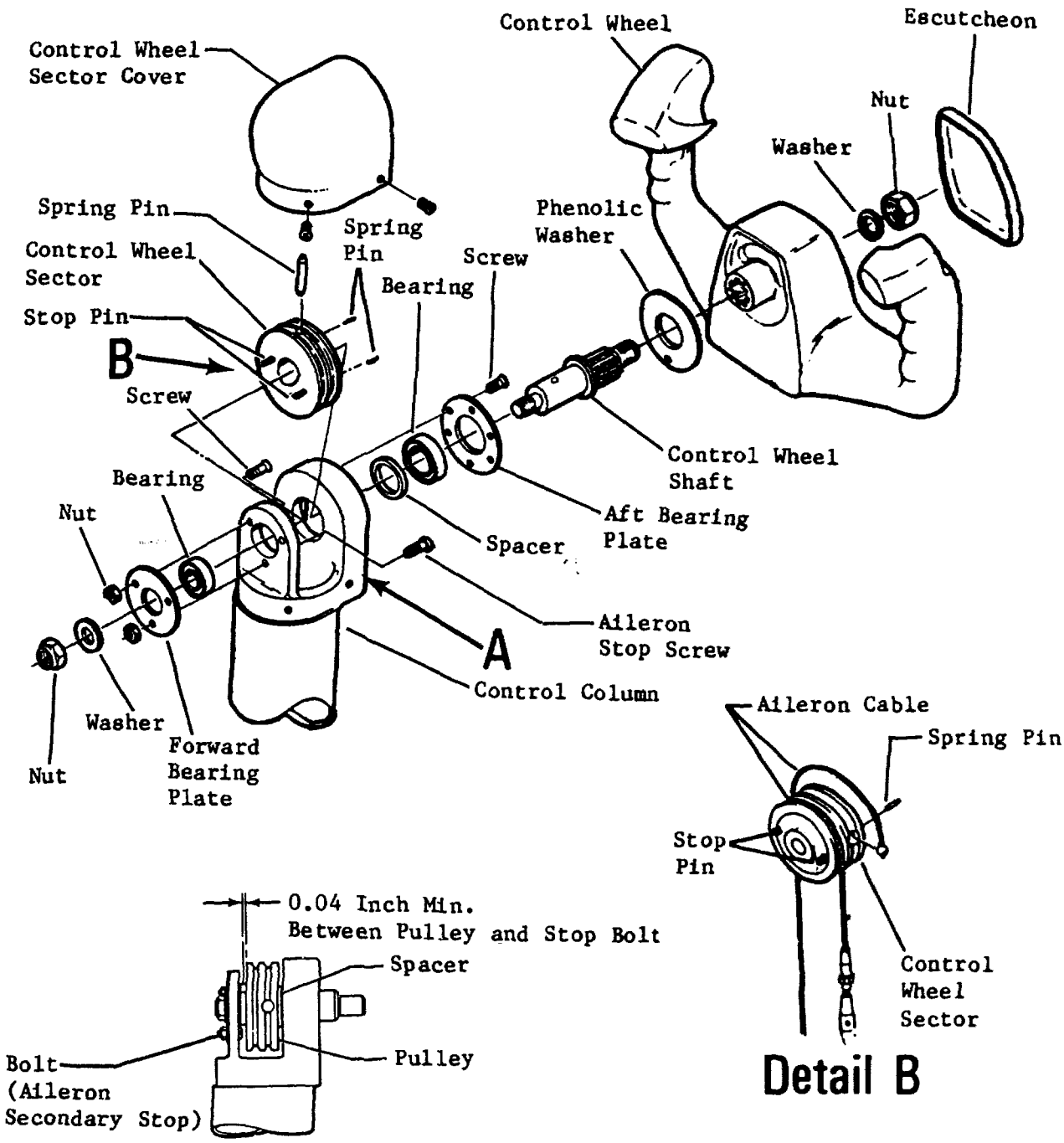
(2) Apply a thin coating of RC680 to outer surface of aft bearing and mating surface of control column fork. Insert bearing in aft control column fork. Wipe off excess compound with a trichloroethane moistened cloth. Allow retaining compound to cure. (Refer to Chapter 20.)
(3) After retaining compound has cured, install aft bearing plate and secure with attaching screws.

CAUTION: WHEN INSTALLING FORWARD BEARING, ENSURE THAT RETAINING COMPOUND IS NOT APPLIED TO INNER SURFACE OF BEARING.

(4) Apply a thin coating of RC680 to outer surface of forward bearing and mating surface of control column fork. Insert bearing in forward control column fork. Wipe off excess compound with a trichloroethane moistened cloth. Allow retaining compound to cure. (Refer to Chapter 20.)

CAUTION: ENSURE THAT AILERON STOP (LARGE SCREW) IS IN PROPER POSITION WHEN INSTALLED.

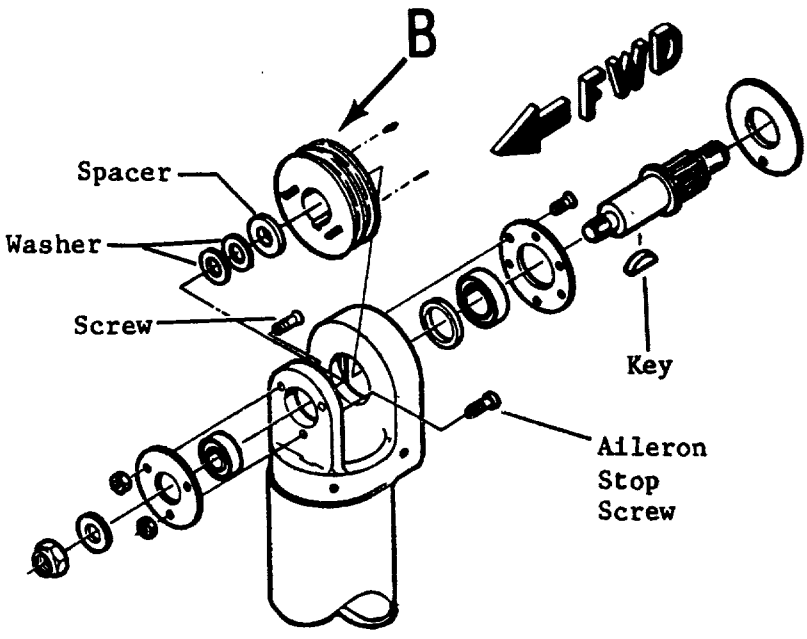
(5) After retaining compound has cured, install forward bearing plate and secure with attaching screws and nuts. Aileron stop (large screw) must be in the lower position.
(6) Prepare aft control column fork for installation of new phenolic washer as follows:
(a) Clean with MEK. Wipe dry with clean dry cloth. Do not allow to air dry as a residue remains.
(b) Repeat step (a) until all old adhesive and contaminants are removed from mating surface.
(c) Brush a uniform coat of 1200 primer on mating surface and allow to air-dry for 30 minutes.
(7) Apply a thin coat of RTV-108 adhesive to phenolic washer and install on control column fork. Ensure that phenolic washer is properly aligned.
(8) Install aileron control cables in control wheel sector and secure with springs.
(9) On *Aircraft 35-018 thru 35-487 and 36-009 thru 36-050*, install key in control wheel shaft, round side in toward center of shaft.
(10) On *Aircraft 35-018 thru 35-487 and 36-009 thru 36-050*, position washers and spacers inside of control column fork. (See Detail A, Figure 201.)
(11) Insert control wheel shaft with index mark on splined shaft at 12 o'clock position and secure on forward side of control column with washer and nut.
(12) On *Aircraft 35-002 thru 35-017 and 36-002 thru 36-008*, install spring pin in control wheel sector.
(13) If control wheel has not been removed from column assembly, proceed with step (14). If control wheel has been removed from control column assembly, proceed as follows:
(a) Route electrical wiring down through control column.



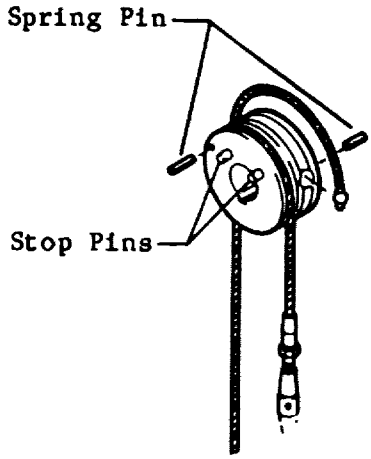
(35-002 thru 35-017, 36-002 thru 36-008 NOT Modified per SSK 947, "Replacement of Control Column Pulley and Shaft Assembly")

8-64B

Control Wheel Installation
Figure 201 (Sheet 1 of 2)

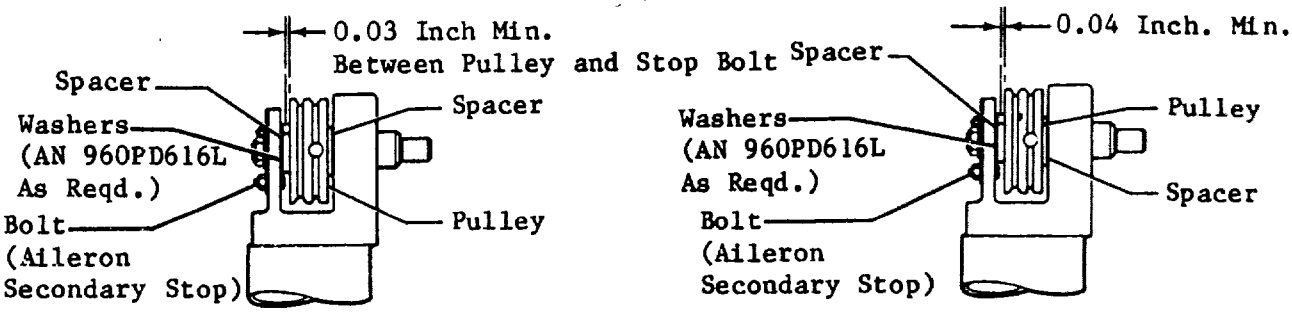


Detail A



Detail B

(35-018 thru 35-487, 36-009 thru 36-050 and Prior Aircraft Modified per SSK 947, "Replacement of Control Column Pulley and Shaft Assembly")



(35-002 thru 35-017, 36-002 thru 36-008 Modified per SSK 947, "Replacement of Control Column Pulley and Shaft Assembly")

(35-018 thru 35-487 36-009 thru 36-050)

Control Wheel Installation
Figure 201 (Sheet 2 of 2)

(b) Install electrical pins into plug P80. (Refer to Chapter 27 of Wiring Manual.)
(c) Connect plug P80 to receptacle J80, located forward of control column base.

(14) When installing control wheel, loop wiring approximately 1-1/2 turns forward of wheel and 1-1/2 turns inside of control wheel hub. Install wheel and ensure that it is in the neutral position.
(15) Secure control wheel with washer and nut. Install control wheel escutcheon.
(16) Install aileron control cables on aileron sector.
(17) Rig and inspect aileron control system. (Refer to 27-10-00.)
(18) Check that control wheels are synchronized in their movement and do not move past control column aileron stop. Verify that wiring is not stressed during rotation of control wheels.
(19) Test all functions of control wheel switches to ensure that all electrical connections are correctly installed.
(20) Install all floorboards previously removed.

C. Remove Control Wheel Assembly. (*Aircraft 35-488 and Subsequent and 36-051 and Subsequent*) (See Figures 202 and 203.)

NOTE: Disassemble control wheel assembly only to extent necessary to complete needed repairs.

(1) Remove control wheel sector cover and control wheel escutcheon.
(2) Remove nut and washer which attach control wheel to shaft and pull control wheel from assembly.
(3) If control wheel is not to be removed further, support control wheel to prevent damage to electrical wiring and proceed to step (4). If control wheel is to be replaced, proceed as follows:
(a) Remove switches from control wheel.
(b) Cut connecting wiring from switches leaving a small amount of coded insulation on soldered end. Retain switches.
(c) Extract wire harness from control wheel.
(4) Remove nut that secures control wheel shaft to control column assembly.
(5) Remove screws which secure aft bearing plate in control column assembly and remove bearing plate.
(6) Remove floorboards to gain access to forward aileron sector and remove forward aileron control cables from forward aileron sector.
(7) Remove shaft, aft bearing, and key from control column.
(8) Remove spring pins which secure control cables in place.
(9) Remove control cables from control wheel sector, tag cables position for proper reinstallation.
(10) Remove control wheel sector and spacers.
(11) Remove screws, nuts, and washers which secure secondary aileron stop and remove secondary aileron stop.
(12) Remove remaining nut and screw which secures forward bearing plate to control column and remove forward bearing plate, laminated washer, washers, and spacer.
(13) Press out forward bearing.

D. Install Control Wheel Assembly. (*Aircraft 35-488 and Subsequent, 36-051 and Subsequent*) (See Figures 202 and 203.)

(1) Clean bearings and mating surfaces of control column with MEK. Wipe dry with clean, dry, cloth. Do not allow to air dry as a residue remains.
(2) Temporarily install forward and aft bearings, sector spacers, washers, laminated washer, forward and aft bearing plates, key, and shaft. Peel laminated washer to shim for 0.005 to 0.012 inch (0.127 to 0.304 millimeter) shaft end play.
(3) Disassemble parts temporarily installed in step (2).

CAUTION: **WHEN INSTALLING FORWARD AND AFT BEARINGS, ENSURE THAT RETAINING COMPOUND IS NOT APPLIED TO THE INNER SURFACES OF THE BEARINGS.**

(4) Apply a thin coating of RC680 to mating surfaces of forward and aft bearings and control column. Press bearings into control column. Wipe off excess compound with a trichloroethane moistened cloth. Allow retaining compound to cure. (Refer to Chapter 20.)

CAUTION: ENSURE THAT RETAINING COMPOUND IS CURED PRIOR TO INSTALLING REMAINING PARTS.

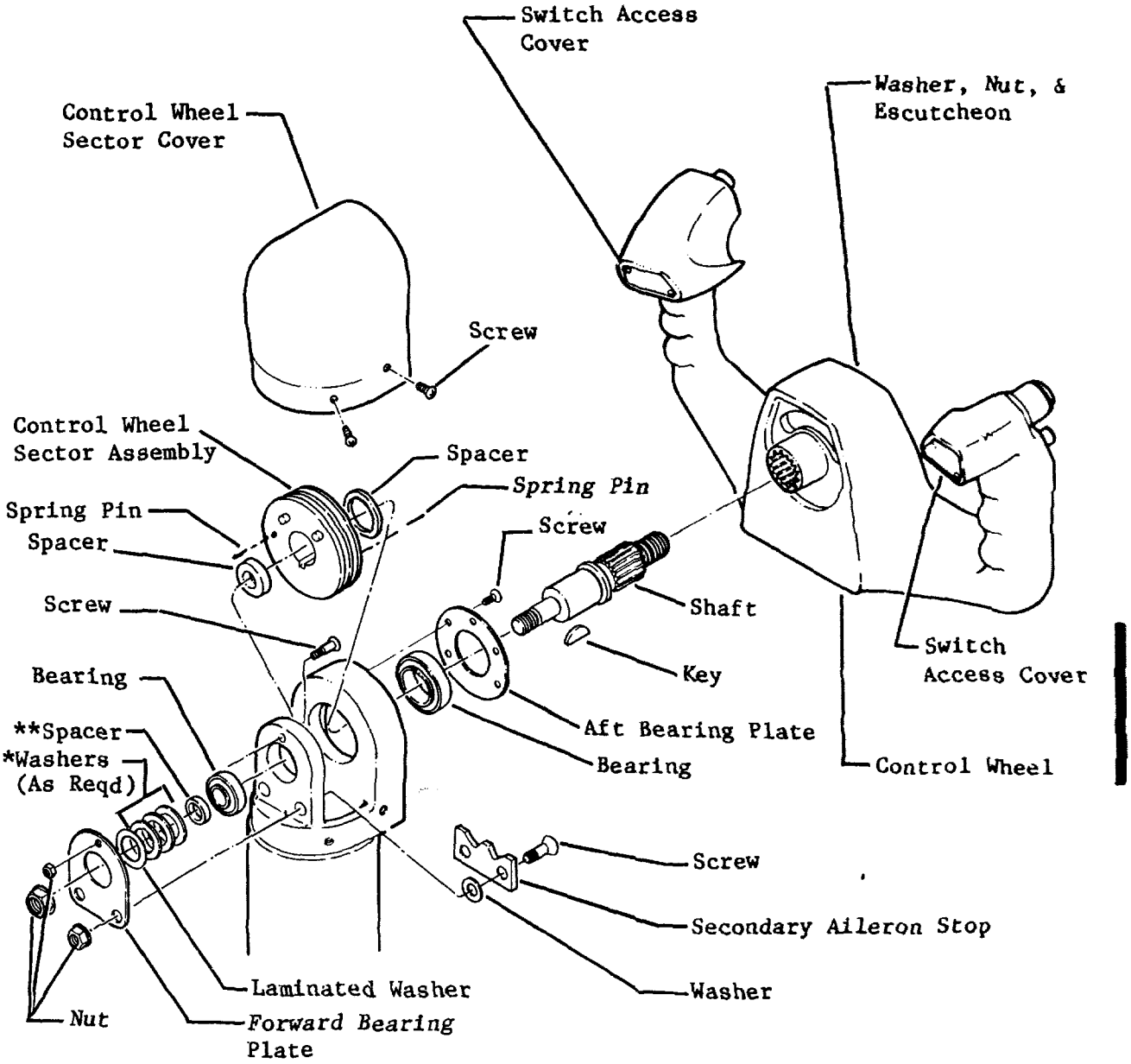
(5) Install aft bearing plate and secure with screws.
(6) Remove tags from aileron control cables, install aileron control cables in control wheel sector, and secure with spring pins.
(7) Install key in shaft, round side in toward center of shaft.
(8) Align spacers and control wheel sector inside of control column fork and insert shaft and key. Index mark on splined shaft shall be at 12 o'clock position.
(9) Place secondary aileron stop and washers in proper position.
(10) Slide spacer, washers, laminated washer, and forward bearing plate over shaft and secure with nut.
(11) Install screws and nuts that secure forward bearing plate and secondary aileron stop.
(12) If control wheel has not been removed from the assembly, proceed with step (13). If control wheel has been removed from control column assembly, proceed as follows:
  (a) Route wire harness into control wheel.
  (b) Note that color code wire ends are in correct locations in control wheel. Refer to wire ends still attached to removed switches.
  (c) Remove wire ends from switches, place heat shrink on harness wire ends, solder wire harness to switches, and install heat shrink tubing over connections.
  (d) Install switches in control wheel.
  (e) Ensure that control wheel and switch access covering mating surfaces are clean and free of dirt, oil, grease, and moisture.
  (f) Apply a thin coat of RC680 to control wheel and switch access cover mating surfaces.
  (g) Position access cover on control wheel and secure with screws. Wipe excess retaining compound off of control wheel. Allow retaining compound to cure. (Refer to Chapter 20.)
(13) When installing control wheel, loop wiring approximately 1-1/2 turns outside of control wheel, then 1-1/2 turns inside of control wheel hub. Install control wheel and ensure that control wheel is in neutral position.
(14) Secure control wheel with washer and nut. Install control wheel escutcheon.
(15) Install aileron control cables on forward aileron sector.
(16) Rig and inspect aileron control system. (Refer to 27-10-00.)
(17) Check that both control wheels are synchronized in their movement and do not move past the control column aileron stop. Verify that wiring is not stressed during rotation of control wheel.
(18) Test all functions of the control wheel switches to ensure that all electrical connections are correctly installed .
(19) Install control wheel sector cover and floorboards previously removed.

D. Remove Control Wheel Switch.

NOTE: Removal and installation procedures are identical for all control wheel switches.

(1) Remove electrical power from aircraft.
(2) Remove set screw securing switch.
(3) Pull switch out of control wheel.

NOTE: It may be necessary to remove escutcheon and unloop wiring from control wheel hub to gain enough slack for removal.



** *(On Aircraft 35-500, 35-502, 35-503, 35-505, 35-509, thru 35-567 and 36-054 thru 36-060, Replace spacer with washer.)*

* Peel laminated washer to shim for 0.005 to 0.012 inch shaft end play prior to applying retaining compound to bearings.

Control Wheel Installation
Figure 202

(4) Disconnect and identify electrical wiring from switch terminals.
(5) Remove switch from aircraft.

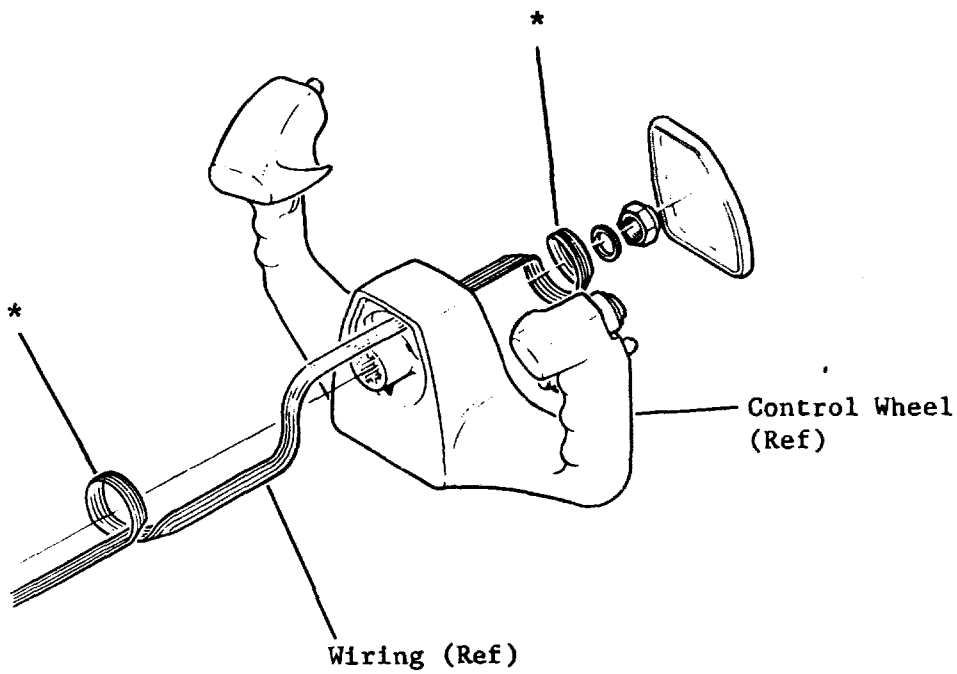
E. Install Control Wheel Switch.
(1) Identify and connect electrical wiring to switch terminals. (Refer to Wiring Manual.)
(2) Install switch into control wheel and secure with set screw.

NOTE: If escutcheon was removed and wiring unlooped, return wiring to configuration shown in figure 203.

(3) Restore electrical power to aircraft.
(4) Perform operational check of applicable system.

* Loop wiring approximately 1-1/2 turns outside control wheel hub, then 1-1/2 turns inside control wheel hub.

Ensure that wiring has sufficient slack when looped so that it will not become stressed when control wheel is rotated to extreme limits.



Control Wheel Installation
Figure 203

AILERON TRIM TAB ACTUATOR - MAINTENANCE PRACTICES

1. TOOLS AND EQUIPMENT

NOTE: Equivalent substitutes may be used in lieu of the following:

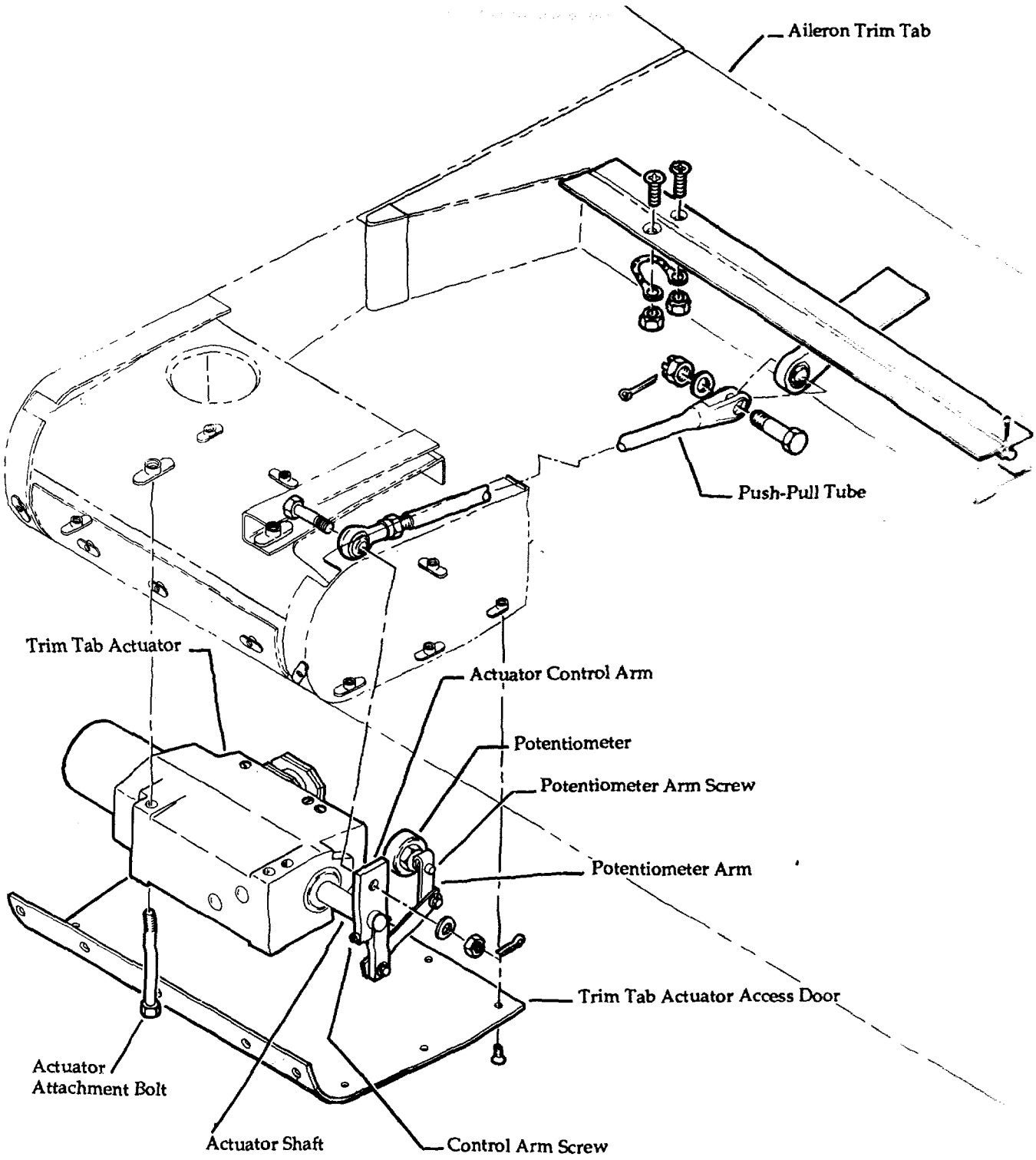
NAME	PART NUMBER	MANUFACTURE	USE
Multimeter	Model 260	Simpson	Resistance Measurements
Masking Tape		Commercially Available	Hold Trim Tab

2. Removal/Installation

NOTE: Access to the trim tab actuator is through the trim tab actuator access door on the aileron leading edge.

A. Remove Aileron Trim Tab Actuator (See figure 201.)
   (1) Disconnect push-pull tube from aileron trim tab.
   (2) Remove trim tab actuator access door from left aileron bottom leading edge.
   (3) Disconnect actuator electrical connector and slide connector back through spar.
   (4) Loosen potentiometer arm screw at potentiometer shaft and remove potentiometer arm from shaft.
   (5) Remove potentiometer and bracket from structure.
   (6) Remove actuator attachment bolts and actuator with push-pull tube attached from aileron.
   (7) Loosen control arm screw and remove actuator control arm from actuator.

B. Install Aileron Trim Tab Actuator (See figure 201.)
   (1) Place actuator control arm on actuator shaft with index spot on arm aligned with index line on actuator shaft. Tighten control arm screw.
   (2) Install actuator with push-pull tube attached in aileron using attaching parts. Connect push-pull tub attached in aileron using attaching parts. Connect push-pull tube to trim tab.
   (3) Install potentiometer and bracket using existing hardware. Tape trim tab in neutral position.
   (4) Using an ohmmeter, rotate potentiometer shaft to center position (±10 ohms). Set potentiometer arm on potentiometer shaft and tighten arm screw.
   (5) Connect electrical connector to actuator and install actuator access door.



Aileron Trim Tab Actuator Installation
Figure 201

4-4C-2

AILERON SECTOR - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Rigging Pin		Local Manufacture	Rigging Controls
Lockwire	MS20995-C41	Commercially Available	Safetying Parts

2. Removal/Installation

NOTE: When aileron sector is removed or replaced, the rigging and inspection of the aileron control system must be performed. (Refer 27-10-00 Adjustment/Test.)

A. Removal Aileron Sector. (See figure 201.)
(1) Remove access covers from copilot's control column, cabin center aisle and fuselage center section, as required to gain access to the control wheel sector and aileron sector.
(2) Insert rigging pin in both control wheel sectors and aileron sector.

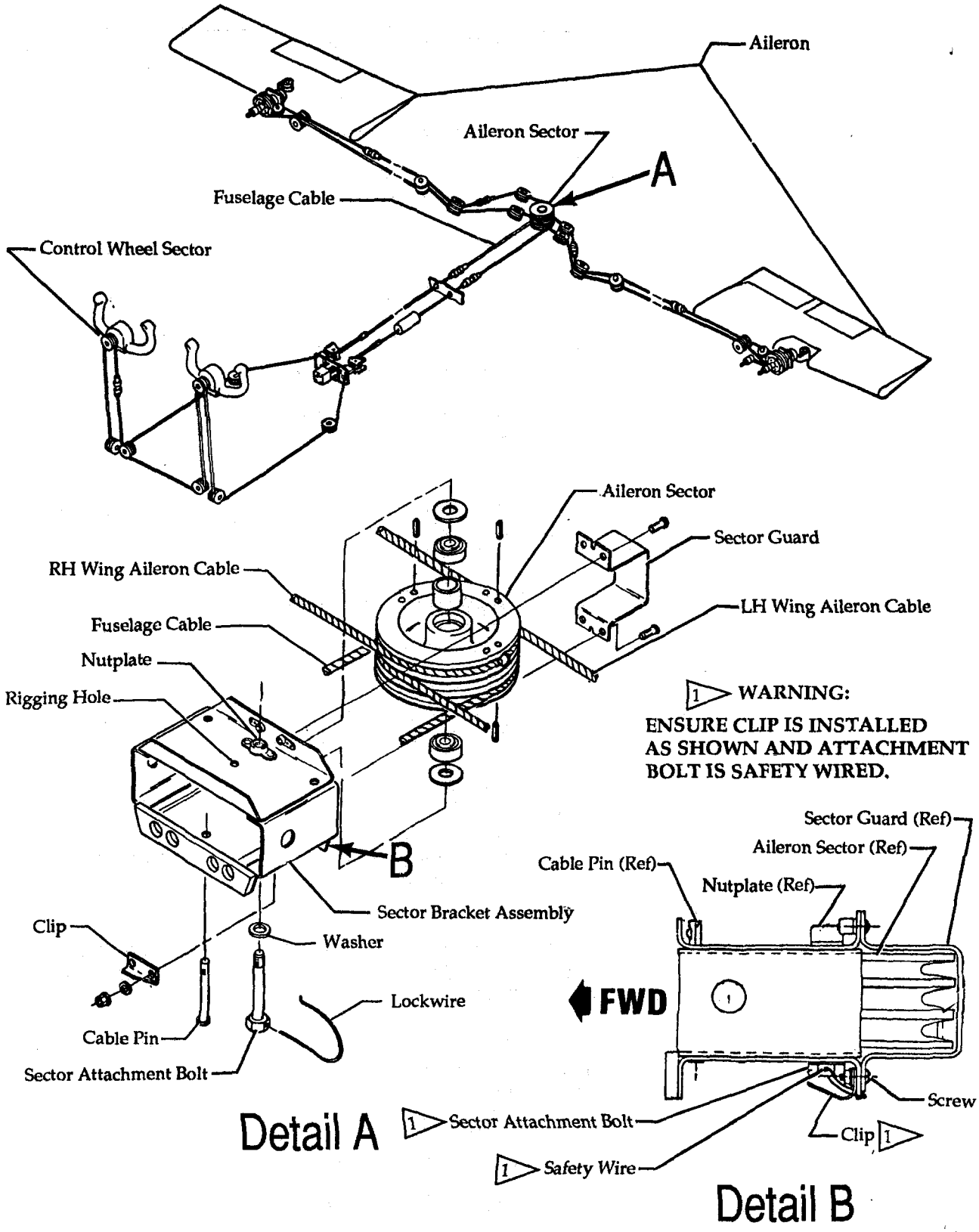
NOTE:
- On *Aircraft 35-108 and Subsequent and 36-032 and Subsequent*, a hole is provided in the keel beam access cover approximately 6 inches forward of frame 22. This hole allows the rigging pin to be inserted in the aileron sector without removal of the access panel.
- Rigging pin is fabricated from 3/16 inch round steel stock. Make 90° bend, approximately 8 1/2 inches from end of 12 inch piece of stock.

(3) Loosen and disconnect wing aileron and fuselage cables from aileron sector.
(4) Remove rigging pin from aileron sector.
(5) Cut safety wire, remove attaching parts, sector guard and clip from aircraft.
(6) Remove sector attachment bolt and washer from bottom side of sector bracket assembly.
(7) Slide aileron sector from sector bracket assembly and remove from aircraft.

B. Install Aileron Sector. (See figure 201.)

WARNING: ENSURE SECTOR ATTACHMENT BOLT IS SAFETIED, AND CLIP IS PROPERLY INSTALLED. THIS WILL PREVENT LOSS OF BOLT AND SUBSEQUENT LOSS OF AILERON CONTROL.

(1) Position aileron sector in sector bracket assembly, install sector attachment bolt, with washer, through sector bracket assembly and secure to nutplate.
(2) Position sector guard and clip on sector bracket assembly and secure with attaching parts.
(3) Safety sector attachment bolt with MS20995-C41 lockwire using double twist method.
(4) Install rigging pin into aileron sector.
(5) Wrap and connect wing aileron and fuselage cables as shown in figure 201.
(6) Remove rigging pins.
(7) Perform rigging aileron control system. (Refer 27-10-00, Adjustment/Test.)
(8) Install all previous equipment removed to gain access to the aileron and control wheel sectors.
(9) Return aircraft to normal.



8-13B-2
A8-9C

Aileron Sector Installation
Figure 201

EFFECTIVITY: ALL

MM-99

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RUDDER AND TAB - DESCRIPTION AND OPERATION

1. DESCRIPTION

A. Directional control of the aircraft is provided by the rudder. The rudder system consists of a dual set of rudder pedals, a cable system, and rudder assembly. The rudder, mounted on a torque tube, is connected to the vertical stabilizer at three hinge points. Rudder control is provided manually by either set of rudder pedals or automatically through the primary or secondary yaw damper servos.

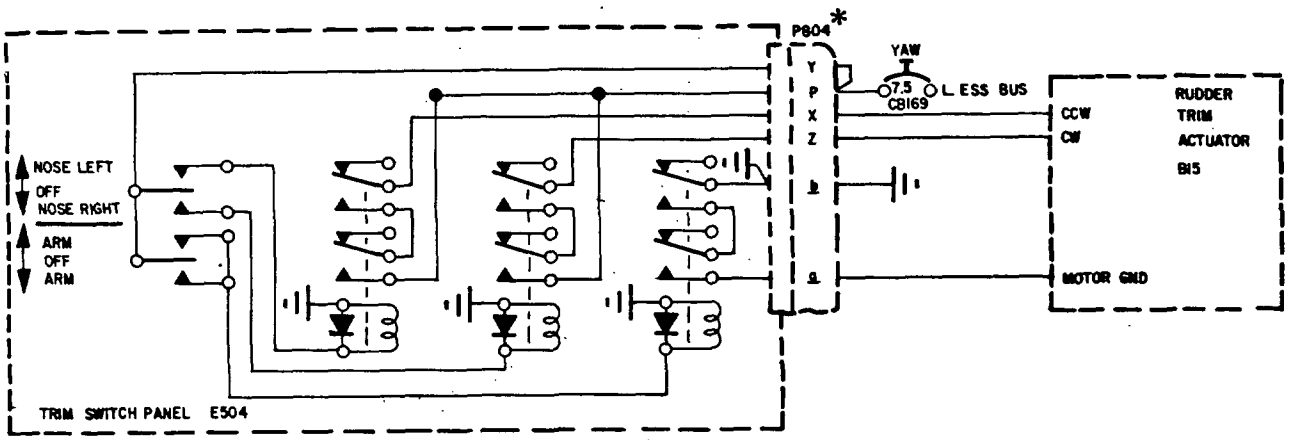
B. A rudder trim tab system is an integral part of the rudder and further increases directional stability of the aircraft. The rudder trim tab is attached to the rudder by a continuous-type hinge and is powered by a rotary-type electrical actuator located in the lower leading edge of the rudder. The Yaw Trim Switch on the cockpit center pedestal controls the tab actuator, and a rudder trim indicator indicates trim tab travel. Trim tab travel is 15° (±1°) left and right.

2. OPERATION

A. Rudder control motion is transferred from interconnected rudder control pedals through push-pull tubes, a bellcrank assembly, then a closed loop drive cable assembly to a drive bellcrank connected to the rudder torque tube. Two sets of rudder travel stop bolts (primary and secondary) are installed. The primary stop bolts, located in the stinger, limit rudder travel when contacted by the rudder torque tube bellcrank. The secondary stop bolts, located on the floorboards forward of the rudder pedals, limit rudder travel when contacted by the rudder pedal arms.

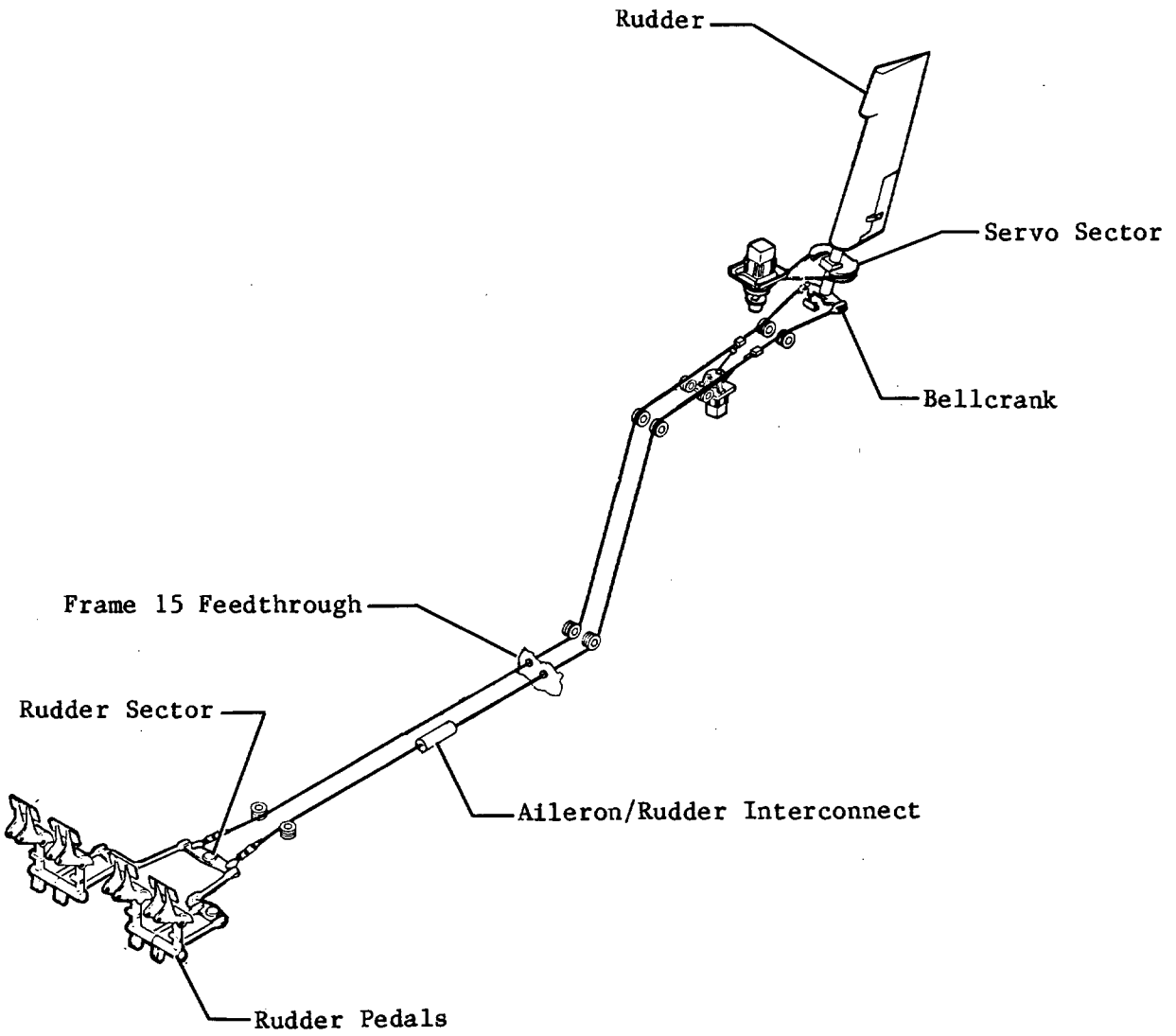
B. Rudder trim tab motion is controlled by the Yaw Trim Switch. Setting the switch to LEFT or RIGHT energizes the trim tab actuator to either clockwise or counterclockwise rotation. A push-pull tube connects the actuator to the trim tab. Maximum trim tab travel is 15° (±1°) both left and right.

C. The rudder trim tab may be manually operated by using the Rudder Trim Switch on the center pedestal. A trim tab position indicator is also installed on the center pedestal.



* Effective Aircraft 35-071 and Subsequent
and 36-019 and Subsequent P804 becomes P2.

Rudder Trim Tab Electrical Control System
Figure 1



Rudder Control System Installation
Figure 2

8-22C-1

RUDDER AND TAB CONTROL SYSTEM - TROUBLE SHOOTING

1. TOOLS AND EQUIPMENT

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Multimeter	Model 260	Simpson	Resistance Checks

2. TROUBLE SHOOTING

A. See figure 101 for rudder control system trouble shooting. See figure 102 for rudder trim control system trouble shooting. (Refer to chapter 27 of wiring manual for wiring diagram.)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
1. Rudder Not in Neutral Position When Rudder Pedals Are Centered.		
a. Rudder control system cables not rigged properly.	Check rudder control system rigging. (Refer to "Adjustment/Test", this section.)	Rig rudder control system. (Refer to "Adjustment/Test", this section.)
b. Primary or secondary stop bolts not adjusted properly.	Check primary stop bolt adjustment.	Adjust primary stop bolt.
2. Rudder Pedal Movement Rough or Excessive Force Required.		
a. Binding in cable control system.	Inspect rudder cables. Check routing, clearances from structure, frozen pulley, and bearings	Reroute cables or repair structure to maintain clearance. Replace cables, pulleys, and bearings as required.
b. Rudder cable tension too great.	Check rudder control system rigging. (Refer to "Adjustment/Test", this section.)	Rig rudder control system. (Refer to "Adjustment/Test", this section.)
c. Rudder rubbing adjacent structure.	Inspect rudder installation and verify clearances. (Refer to 27-20-01.)	Reinstall rudder, if necessary. (Refer to 27-20-01.)

Rudder Control System Trouble Shooting
Figure 101 (Sheet 1 of 2)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
3. Excessive Free Play in Rudder Pedals.		
a. Rudder pedals rigged incorrectly.	Check freeplay of forward push-pull tubes and aft tubes. (Refer to "Adjustment/Test this section.)	Rig rudder control system. (Refer to "Adjustment/Test", this section.)
b. Rudder cable tension too low.	Check rudder control system rigging. (Refer to "Adjustment/ Test", this section.)	Rig rudder control system. (Refer to "Adjustment/test", this section.)

Rudder Control System Trouble Shooting
Figure 101 (Sheet 2 of 2)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
1. No Yaw Trim When Yaw Trim Switch is Actuated.		
a. Loss of 28 vdc power to yaw trim circuit.	Check that YAW circuit is depressed.	Reset circuit breaker.
	Check for 28 vdc between P36, pin A and aircraft ground with Yaw Trim Switch set for nose right. (Refer to wiring diagram 27-20-03.) If 28 vdc is not present, refer to step b.	If 28 vdc is present, check wiring to yaw trim actuator. Replace or repair defective wiring. If wiring is OK, replace yaw trim actuator.
	Check for 28 vdc between P36, pin B and aircraft ground with Yaw Trim Switch set for nose left. If 28 vdc is not present, refer to step b.	If 28 vdc is present, check wiring to yaw trim actuator. Replace or repair defective wiring. If wiring is OK, replace yaw trim actuator.
b. Yaw Trim Switch malfunction.	Check for 28 vdc between P804, pin Z and aircraft ground with Yaw Trim Switch set for nose right. (Refer to wiring diagram 27-20-03.)	If 28 vdc is not present, replace trim switch panel (E504).
	Check for 28 vdc between P804, pin X and aircraft ground with Yaw Trim Switch set for nose left.	If 28 vdc is not present, replace trim switch panel (E504).
c. Rudder trim tab actuator malfunction.	Verify 28 vdc power to actuator. (Refer to wiring diagram 27-20-03.)	If 28 vdc is present, replace rudder trim tab actuator.
d. Mechanical jam or failure.	Inspect rudder trim tab installation for proper clearance. Verify that push-pull tube and actuator control arm are secure and clear of adjacent structure.	Reinstall trim tab, or repair or replace damaged parts. (Refer to 27-20-03.)

Rudder Trim Control System Trouble Shooting
Figure 102 (Sheet 1 of 2)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
2. No Yaw Trim Indication With Trim Actuator Moving For Nose Left or Nose Right.		
a. Yaw trim indicator malfunction.	Check that TAB & FLAP POS circuit breaker is depressed.	Reset circuit breaker.
	Check for 0.1 vdc (± 10%) between P36, pin C and aircraft ground with rudder trim tab set for nose right. (Refer to wiring diagram 27-20-03.)	If 0.1 vdc (nose right) and 0.1 vdc (nose left) is present or or resistance is not within tolerance, replace roll trim indicator.
	Check for 0.1 vdc (±10%) between P36, pin C and aircraft ground with rudder trim tab set for nose left.	
	Disconnect P754 from indicator and check resistance from pin A to pin C. Resistance should be 100 Ω ±10%).	

Rudder Trim Control System Trouble Shooting
Figure 102 (Sheet 2 of 2)

RUDDER CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Rudder and Tab Rigging Fixture	2471005	Learjet Inc. Wichita, KS	Measure degree of control deflection.
Push-Pull Spring Scale (0-10 lb [0-4.54 kg])		Commercially Available	Measure force.
Tensiometer		Commercially Available	Measure cable tension.
Tensiometer Adapter*	2471007	Learjet Inc. Wichita, KS	Measure cable tension.

* The tensiometer adapter is intended for use on servo cables in areas that are inaccessible to normal tensiometer usage.

2. Adjustment/Test

NOTE: Prior to rigging rudder control system, perform operational check of rudder control system to ensure proper cable installation. (Refer to Inspection/Check, this section.)

A. Rig Rudder Control System

(1) Remove access covers and floorboards as required for access to rudder control system.

(2) Insert rigging pin in left and right rudder pedal bellcranks. This provides a stationary point from which the rudder control system can be rigged.

(3) Adjust rudder pedal bellcrank push-pull tubes until centerline of rudder pedal supports are 7 (±1)° aft of vertical.

(4) Adjust rudder sector push-pull tubes until rudder sector bellcrank is perpendicular to BL 0.00.

(5) Determine that the rudder is in neutral position by use of contour plates or rudder and tab rigging fixture. If rudder is not in neutral, adjust cable turnbuckles until rudder is in neutral.

NOTE: When cable turnbuckles are loosened or tightened to adjust cable tension, it may be necessary to readjust the rudder to the neutral position. Repeat cable adjustments alternately until rudder is rigged to neutral at the required cable tension.

(6) Adjust rudder control cable tension in accordance with procedures outlined in step 2.D.

(7) Remove rigging pins.

(8) Using rudder and tab rigging fixture (P/N 2471005) or see Figure 201 if fixture is not available, adjust the primary travel stops in the tailcone to obtain a rudder travel as shown in Figure 201.

(9) Lock primary stop bolts with jamnuts. On *Aircraft 35-040 and Subsequent and 36-017 and Subsequent and earlier aircraft modified by SB 35/36-27-3,* safety wire stop bolts.

(10) Adjust the secondary travel stops to contact when 100 (±25) pounds of pressure is applied to the rudder pedals. Lock stop bolts with jamnuts. On *Aircraft 35-040 and Subsequent and 36-017 and Subsequent and earlier aircraft modified by SB 35/36-27-3,* safety wire stop bolts.

(11) Adjust primary yaw damper servo cable tension to 25 (±5) pounds at 75°F. (Refer to 27-00-00, Adjustment/Test.)
(12) Operate rudder several times to ensure proper travel and freedom of movement.
(13) Perform the rudder control system friction test (with yaw damper and autopilot disengaged):
  (a) With the aileron and rudder systems centered, place a push-pull spring scale (accurate within 1/2 pound) at the lower aft corner of the rudder trim tab.
  (b) Slowly apply a horizontal force to the rudder trim tab.
  (c) When the rudder starts its normal movement, read the force applied. The maximum allowable force is 6.5 pounds.
  (d) If the force applied is greater than 6.5 pounds; check for proper cable tensions; check pulleys, sector, and bellcrank for freedom of movement, inspect system for binding; check system for proper lubrication; and ensure that the aileron and rudder control systems are properly rigged.
  (e) After all discrepancies have been corrected, perform steps (a) thru (c) and ensure that the force applied is less than 6.5 pounds.
(14) Check capstan slip clutch adjustment. (Refer to Chapter 22.)
(15) Install and secure all access covers and floorboards prior to flying aircraft.
(16) Restore aircraft to normal.

B. Rudder Trim Tab Travel
(1) Set Battery Switches on.
(2) Set Inverter Switches on.
(3) Set and hold Rudder Trim Switch, located on trim switch panel, to position rudder trim tab to indicator neutral position and attach protractor to rudder and record degrees.

NOTE: Rudder trim tab may not be streamlined with the rudder at indicator neutral. Check aircraft log book for rudder trim tab rod end turns or if the streamlined position of the rudder trim tab is acceptable for flight neutral.

(4) Set and hold Rudder Trim Switch, located on trim switch panel, to LEFT. Trim tab shall travel 15 (±1)° and stop. Record degrees traveled from rudder trim tab indicator neutral.
(5) Set and hold Rudder Trim Switch, located on trim switch panel, to RIGHT. Trim tab shall travel 15 (±1)° and stop. Record degrees traveled from rudder trim tab indicator neutral.
(6) Cycle rudder trim tab several times to check for proper operation and travel. Check rudder trim tab travels from stop to stop in 5 to 20 seconds.
(7) Set and hold Rudder Trim Switch, located on trim switch panel, to position rudder trim tab to indicator neutral position
(8) Remove protractor from rudder.
(9) Set Battery and Inverter Switches off.
(10) Restore aircraft to normal.

C. Rig Rudder Trim Tab
(1) Set Battery Switches on.
(2) Set Inverter Switches on.
(3) Set and hold Rudder Trim Switch, located on trim switch panel, to position rudder trim tab streamline with rudder.
(4) Mark centerline of both the rudder and rudder trim tab at the tab/rudder intersection.
(5) Set and hold Rudder Trim Switch, located on trim switch panel, to LEFT to obtain full left travel.
(6) Measure between rudder and tab centerlines and record.
(7) Set and hold Rudder Trim Switch, located on trim switch panel, to RIGHT to obtain full right travel.
(8) Measure between rudder and tab centerlines and record.
(9) Add measurements obtained in step (6) and (8) together and divide by 2.

NOTE: The total left travel measurement plus right travel measurement must fall between 2.5 and 2.9 inches.

(10) Set and hold Rudder Trim Switch, located on trim switch panel, to LEFT and trim rudder trim tab back to the amount of the measurement obtained in step (9).
(11) Verify rudder trim tab is positioned at neutral and the rudder trim position indicator is centered ±1 needle width.
(12) Cycle trim tab several times to check for proper operation and travel. Check that trim tab travels from stop to stop in 5 to 20 seconds.
(13) Set Battery and Inverter Switches off.
(14) Restore aircraft to normal.

D. Adjust Rudder Control Cable Tension

NOTE:
- Prior to rudder control cable tension adjustment, torque wrench tensiometer adapter calibration must be accomplished. (Refer to Adjustment/Test, 27-00-00.)
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The Cable Tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to Adjustment/Test, 27-00-00.)

(1) Remove aft pedestal assembly. (Refer to Chapter 53.)
(2) Remove applicable carpeting and floorboards, as required, to gain access to rudder forward sector assembly.
(3) With rudder in faired position, insert rigging pin in left and right rudder pedal bellcranks.
(4) Open tailcone access door, allowing access to rudder control cables and turnbuckles.
(5) Remove locking clips from turnbuckles.
(6) If a new cable is being installed, perform the following:
  (a) Adjust rudder cable tension to 200 (±10) pounds (90.7 [±4.5] kg).
  (b) Cycle rudder system 25 times to seat in and pre-stretch new cable.
  (c) Reduce cable tension.
(7) Adjust rudder control cable tension to 75 (±5) pounds (34 [±2.2] kg).

NOTE:
- The threads at opposite ends of the turnbuckle barrel shall be started at the same time so thread engagement on both ends will be approximately equal.
- When cable turnbuckles are loosened or tightened to adjust cable tension, it may be necessary to readjust the rudder to the neutral position. Repeat cable adjustments alternately until rudder is rigged to neutral at the required cable tension.
- On *Aircraft with lead-acid batteries*, apply corrosion protection to control cable in area of aircraft batteries. (Refer to 27-00-00, Cleaning/Painting.)
- The final adjusted turnbuckle combination shall have no more than three threads of the terminal, fork, or eye exposed outside the barrel of the turnbuckle.

(8) Once proper cable tension has been obtained, remove rigging pin and operationally check rudder control system. (Refer to Inspection/Check, this section.)
(9) Install locking clips in turnbuckles.

(10) Install all previously removed access covers, floorboards, and carpeting.
(11) Install aft pedestal assembly. (Refer to Chapter 53.)
(12) Restore aircraft to normal.

3. Inspection/Check

A. Check Rudder for Proper Operation
(1) During operational check, observe response and freedom of movement of the rudder pedals.
(2) Apply force to pilot's outboard rudder pedal; observe rudder trailing edge movement to the left.
(3) Apply force to pilot's inboard rudder pedal; observe rudder trailing edge movement to the right.
(4) With autopilot engaged, relax force applied to rudder pedals; observe rudder movement to the neutral position.
(5) Observe response and freedom of movement of the rudder pedals.
(6) Apply force to copilot's outboard rudder pedal; observe rudder trailing edge movement to the right.
(7) Apply force to copilot's inboard rudder pedal; observe rudder trailing edge movement to the left.
(8) With autopilot engaged, relax force applied to rudder pedals; observe rudder movement to the neutral position.

B. Check Rudder Trim Tab for Proper Operation
(1) Set Yaw Trim Switch to the RIGHT position and hold in that position until trim tab has reached its travel extreme.

NOTE: The trim tab actuator will continue to run until the switch lever is released or until the trim tab reaches a travel extreme that is controlled by internal stops of the actuator.

(2) The trim tab trailing edge will move to the left in a smooth, responsive movement.
(3) The trim indicator on the center pedestal will indicate corresponding trim tab travel.
(4) Set the Yaw Trim Switch to the LEFT position and hold in that position until trim tab has reached its travel extreme.
(5) The trim tab trailing edge will move to the right in a smooth, responsive movement.
(6) The rudder trim indicator will indicate corresponding trim tab travel.

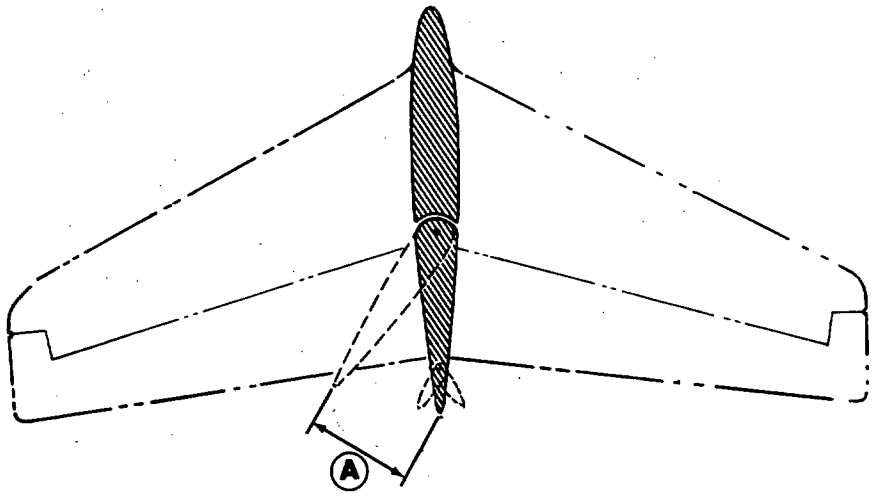
C. Perform Rudder Control Cable Tension Check

NOTE:
- Perform rudder control cable tension check in accordance with the current inspection intervals specified in Chapter 5.
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The cable tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to 27-00-00, Adjustment/Test.)

(1) Position rudder to the neutral position.
(2) Open tailcone access door and attach tensiometer to rudder control cables. Rudder cable tension shall be 75 (±5) pounds at 75°F (34 [±2.2] kg at 24°C).

NOTE:
- The rudder cable must be lying against the back of the tensiometer when pressure is applied so that no extra pull is applied to the cable.
- Avoid pushing or pulling on the tensiometer or cable while a reading is being taken.

(3) Adjust Rudder Control Cable Tension if readings do not fall within tolerances. (Refer to Adjustment/Check, this section.)
(4) Remove tensiometer from rudder control cable and close tailcone access door.
(5) Restore aircraft to normal.



RUDDER TRAVEL CHECK:

1. Set rudder trim tab to neutral (streamlined with rudder).
2. Set rudder to neutral (align lower tip of rudder trim tab with point of stinger).
3. Press left rudder pedal until rudder horn contacts rudder stop bolt, then measure dimension A (distance between lower tip of trim tab and stinger point.)
4. Dimension A should be between 10.74 to 11.83 inches (30° +2°, -1°).
5. Repeat procedure for right rudder travel.

Rudder Rigging
Figure 201

RUDDER - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Tensiometer		Commercially Available	Tensioning Cables
Cable Clamps		Local Manufacture	Clamping Cables

B. Remove Rudder (See figure 201.)

NOTE: Access to the torque tube hinge (lower hinge point) bolt is through access covers on either side of the tailcone stinger. Access to the upper and middle rudder hinge points is through access covers on the rudder leading edge.

(1) Remove screws from upper and lower access covers on rudder leading edge.
(2) Deflect rudder to one side and remove access covers.
(3) Remove rudder servo sector and bellcrank access covers from tailcone.
(4) Disconnect bonding jumper from rudder.
(5) Disconnect trim tab actuator electrical connector and remove wire bundle clamp from rib.
(6) Remove bolt and clip from left side of servo sector and remove cable.
(7) Remove both nuts on forward side of sector which hold sector to torque tube. Remove sector with turnbuckle end of cable attached to sector.

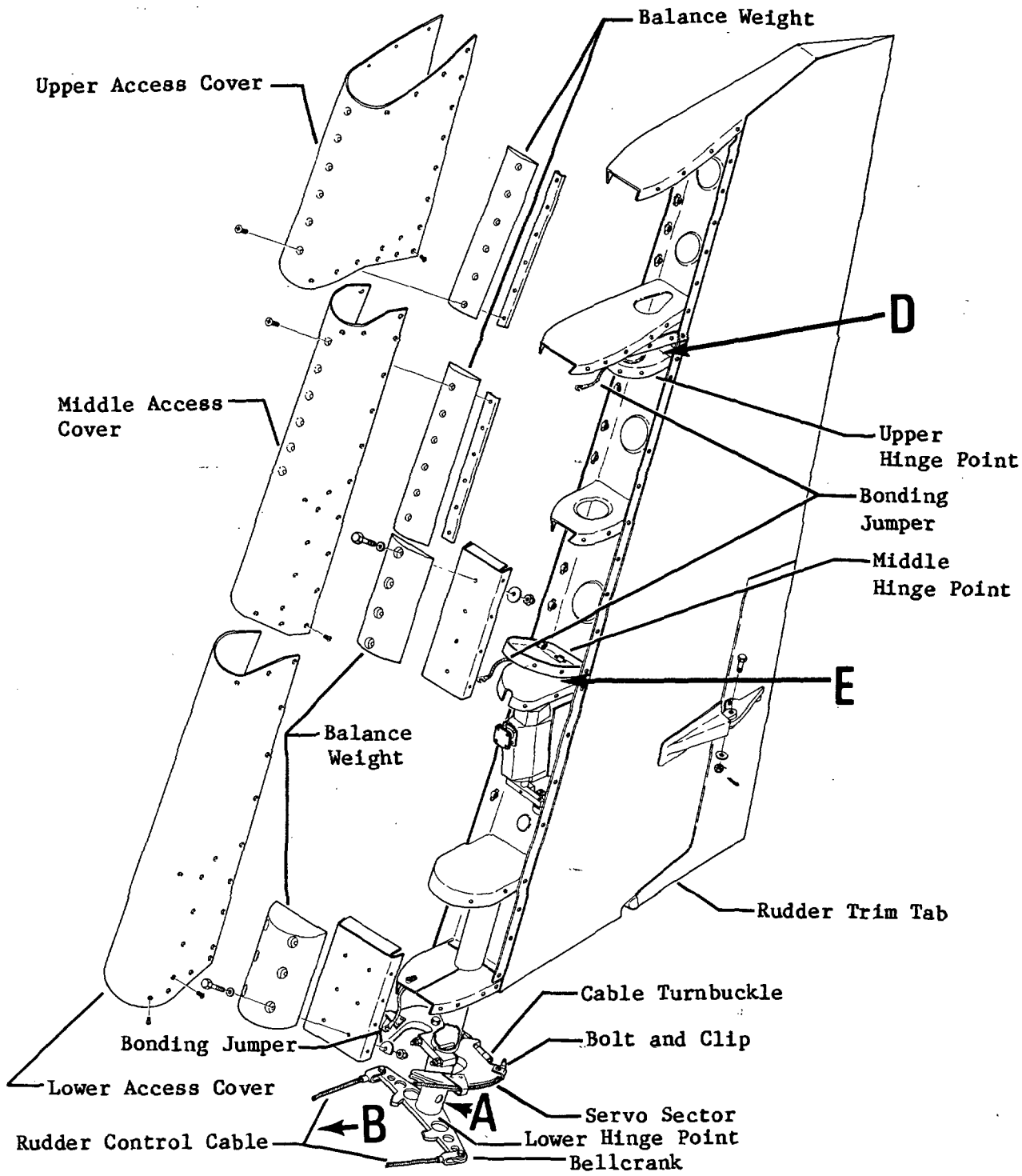
CAUTION: WHEN CLAMPING RUDDER CABLES DO NOT OVERTIGHTEN CLAMPS. OVERTIGHTENING CLAMPS COULD CAUSE CABLE DAMAGE.

(8) Clamp rudder control cables where they pass through bulkhead in a manner such that cables will remain tight forward of bulkhead (see Detail A).
(9) Deflect rudder so that one rudder control cable becomes slack; remove that bolt and cable end from bellcrank. Remove opposite cable in same manner.
(10) Remove torque tube hinge fitting bolt at bottom of rudder torque tube.
(11) Remove middle and upper rudder hinge bolts and washers.
(12) Deflect rudder so that it clears aft bullet navigation light fairing and remove rudder.

C. Install Rudder (See figure 201.)

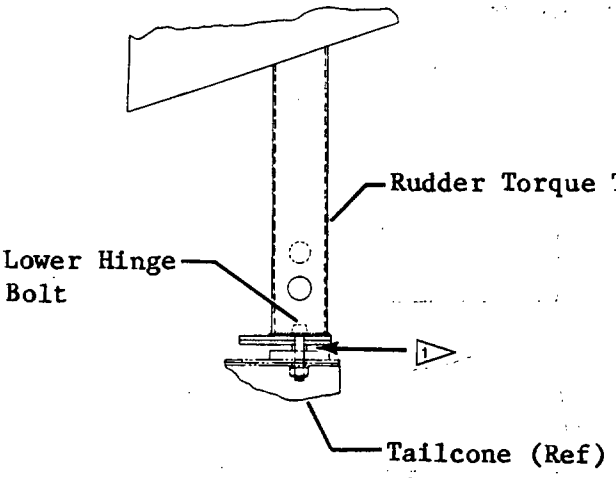
NOTE: If any portion of a control surface is repainted, repaired, or replaced, a check must be made to determine if the balance is still within the prescribed limits. (Refer to 27-00-00, Adjustment/Test.)

(1) Position rudder on vertical stabilizer.
(2) Insert washers and bolts at upper and middle hinge points.
(3) Attach torque tube to lower hinge fitting point and attach bonding jumper.
(4) Deflect rudder to one side and attach rudder control cable on that side. Attach opposite cable in same manner. Remove clamps from rudder control cables.
(5) Attach servo sector to torque tube. Attach cable end to servo sector with bolt and clip.



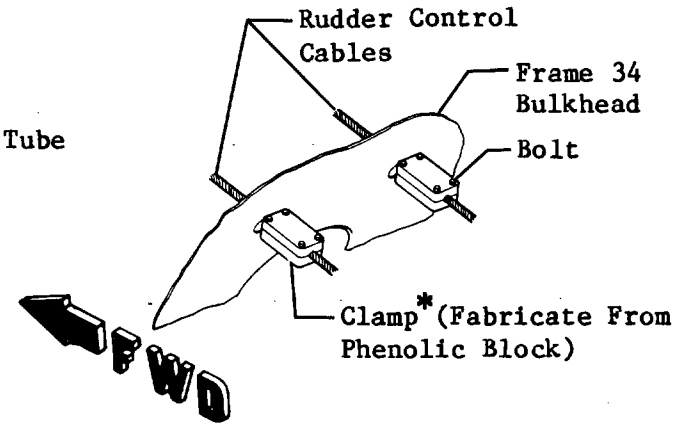
S-13D

Rudder Installation
Figure 201 (Sheet 1 of 3)



Detail A

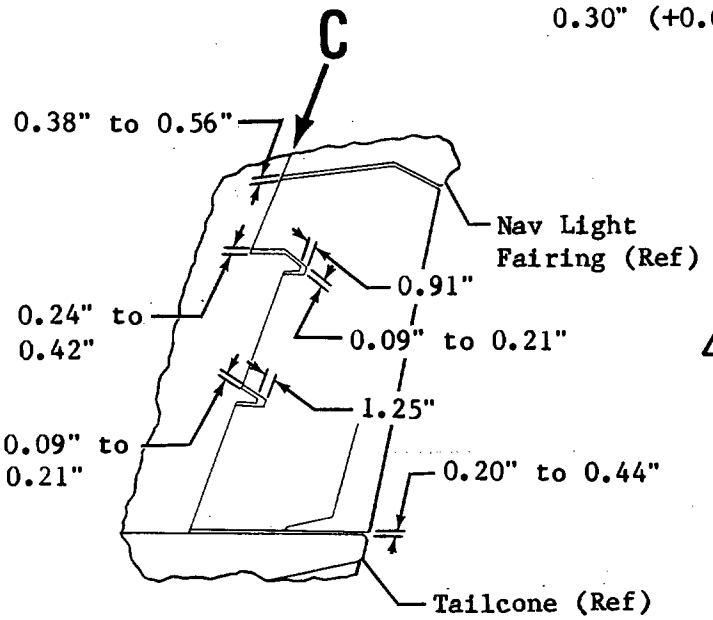
1 Shim with AN 960 washers if and as required to mate bearing and rudder assembly.

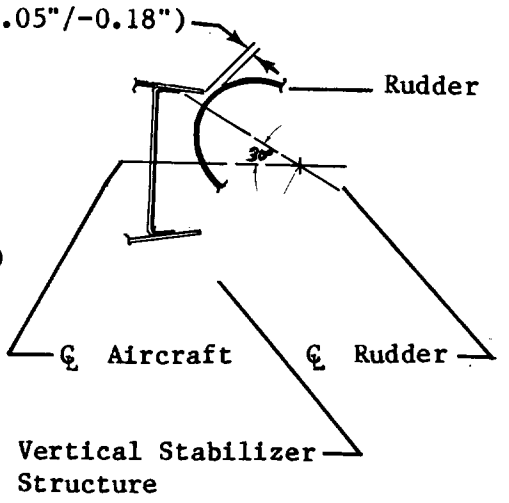


(RUDDER CONTROL CABLE CLAMPING)

* CAUTION: OVERTIGHTENING OF CLAMP COULD CAUSE CABLE DAMAGE.

Detail B

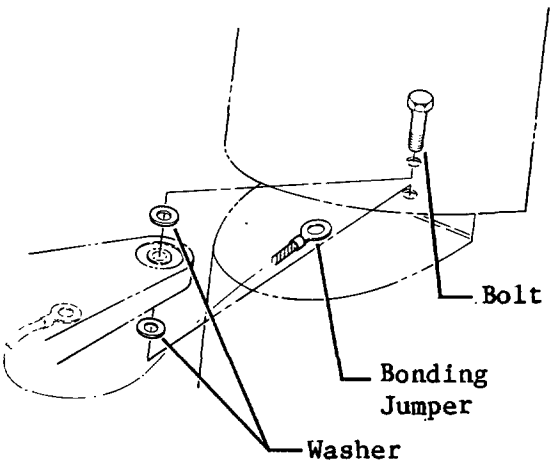




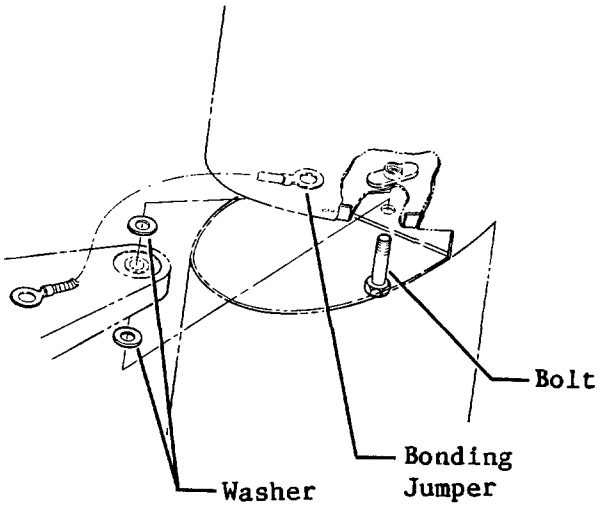
Detail C

(RUDDER CLEARANCES)

Rudder Installation
Figure 201 (Sheet 2 of 3)



Detail D



Detail E

DE5-1C

Rudder Installation
Figure 201 (Sheet 3 of 3)

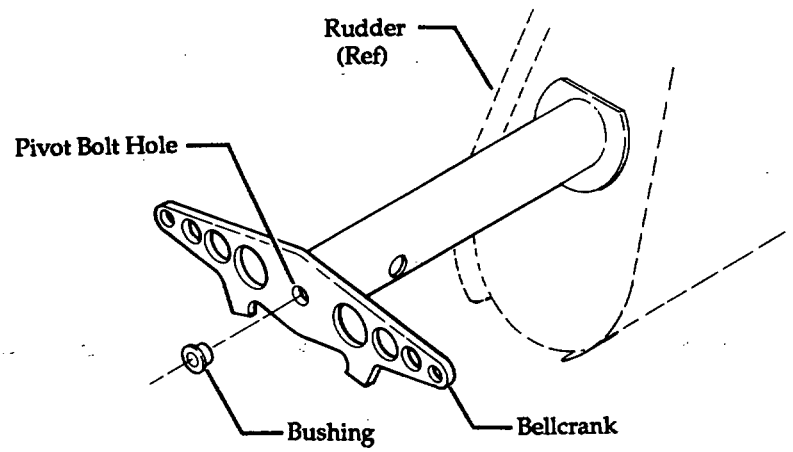
(6) Adjust servo cable to 25 (±5) pounds at 75° F. (Refer to 27-00-00, Adjustment/Test.)
(7) Adjust rudder control cables to 75 (±5) pounds at 75° F. (Refer to 27-00-00, Adjustment/Test.)
(8) Ensure that trim tab actuator wire bundle is wrapped approximately 1-1/2 turns around the torque tube and tied so that the wire will not fall into the bellcrank stops.
(9) Connect trim tab actuator electrical connector and secure wire bundle clamp to rib.
(10) Check rudder control system for proper rigging. (Refer to 27-20-00, Adjustment/Test.)
(11) Install upper and lower access covers.

2. Approved Repairs
A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Bushing	2384033-2	Learjet Inc. Wichita, KS	Pivot hole repair
Epoxy Primer	Refer to Chapter 20 for part number and manufacturer		Bushing installation

B. Repair of Rudder Assembly Pivot Bolt Hole. (See figure 202.)
(1) Remove rudder from aircraft. (Refer to Removal/Installation, this section.)
(2) Ream pivot bolt hole hole in rudder bellcrank to .3465 (+.0010, -.0000) inches.
(3) Install bushing using wet epoxy primer.
(4) Install rudder. (Refer to Removal/Installation, this section.)



Rudder Repair
Figure 202

RUDDER SERVO SECTOR - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Tensiometer		Commercially Available	Tensioning Cables
Rigging Pin		Locally Fabricate	Rig Rudder

2. Removal/Installation

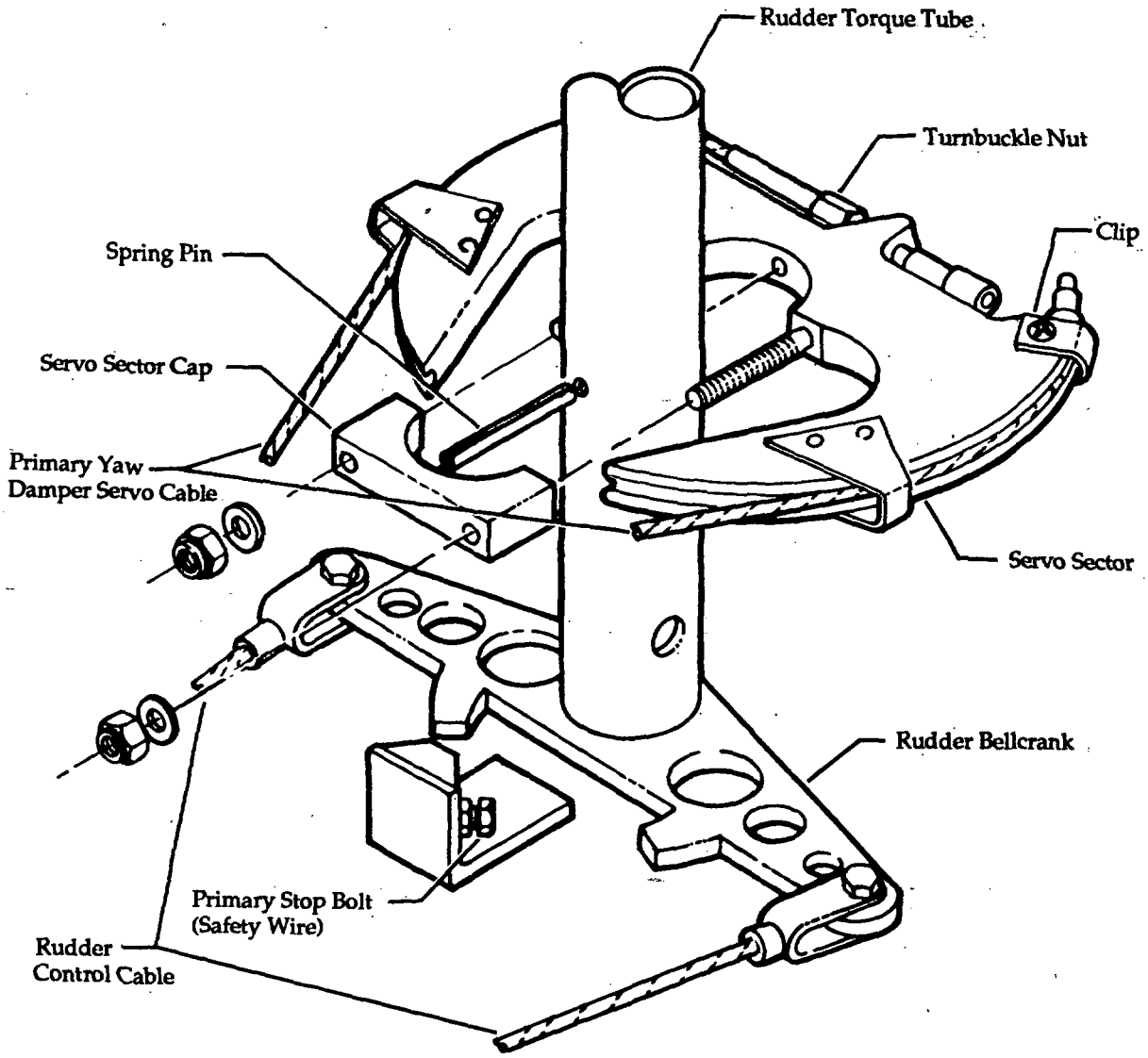
NOTE:
- The autopilot primary yaw damper servo is connected to the rudder servo sector.
- Removal and installation of the rudder servo sector is necessary when a new rudder is to be installed, therefore it has been covered in this chapter.

A. Remove Rudder Servo Sector (See figure 201.)

NOTE: Rigging pin is fabricated from 3/16 inch (4.76 millimeter) round steel stock. Make 90° bend, approximately 8 1/2 inches (21 centimeters) from end of 12 inch (30.5 centimeters) piece of stock. Paint short end of piece red (for visibility).

(1) Remove floorboards as required to gain access to rudder pedal bellcrank. Insert rigging pin in rudder pedal bellcrank.
(2) Remove rudder servo sector and bellcrank access cover from tailcone.
(3) Remove primary yaw damper servo cable turnbuckle nut, clip, and cotter pin, releasing yaw damper servo cable ends from rudder servo sector.
(4) Remove nuts and bolts from servo sector, releasing servo sector and cap from rudder torque tube.

B. Install Rudder Servo Sector (See figure 201.)
(1) Align servo sector and cap with servo sector spring pin in rudder torque tube. Secure servo sector and cap on torque tube using existing hardware.
(2) Attach primary yaw damper servo cable to servo sector using clip, nut, and cotter pin.
(3) Adjust servo cable to 25 (±5) pounds at 75°F. (Refer to 27-00-00 for cable tension temperature-load correction.)
(4) Install rudder servo sector and bellcrank access cover on tailcone.
(5) Remove rigging pin from rudder pedal bellcrank and install floorboards.



8-39A-6

Rudder Servo Sector Installation
Figure 201

RUDDER TRIM TAB - MAINTENANCE PRACTICES

1. Removal/Installation
   A. Remove Rudder Trim Tab. (See Figure 201.)
      (1) Remove rudder from aircraft. (Refer to 27-20-01, Removal/Installation.)
      (2) Disconnect trim tab actuator push-pull tube and bonding jumper from trim tab and trim tab actuator.
      (3) Using a No. 30 (0.128 inch, 3.25 millimeter) bit, drill out tab hinge rivets from rudder. Remove rudder trim tab from rudder.
   B. Install Rudder Trim Tab. (See Figure 201.)
      (1) Install trim tab on rudder with trim tab hinge between rudder spar flange and skin of rudder. Adjust trim tab for proper clearances between rudder tab and rudder as shown. (See Figure 201.)
      (2) Place a straightedge on rudder trailing edge and adjust rudder trim tab until both the rudder and rudder trim tab trailing edges are aligned.
      (3) With trim tab properly positioned, mark rivet locations at upper and lower ends of trim tab. Remove trim tab from rudder and drill a No. 30 hole at each location.
      (4) Install trim tab and cleco in place. Using remaining holes in rudder as a pattern, drill a No. 30 hole at each location except the second from the bottom. Drill a No. 19 (0.166 inch, 4.22 millimeter) hole at this location for bonding jumper attachment.
      (5) Secure trim tab to rudder with rivets (P/N MS20426AD4) and connect bonding jumper to trim tab.
      (6) Install push-pull tube.
      (7) Install rudder. (Refer to 27-20-01, Removal/Installation.)

2. Inspection/Check
   A. Tools and Equipment

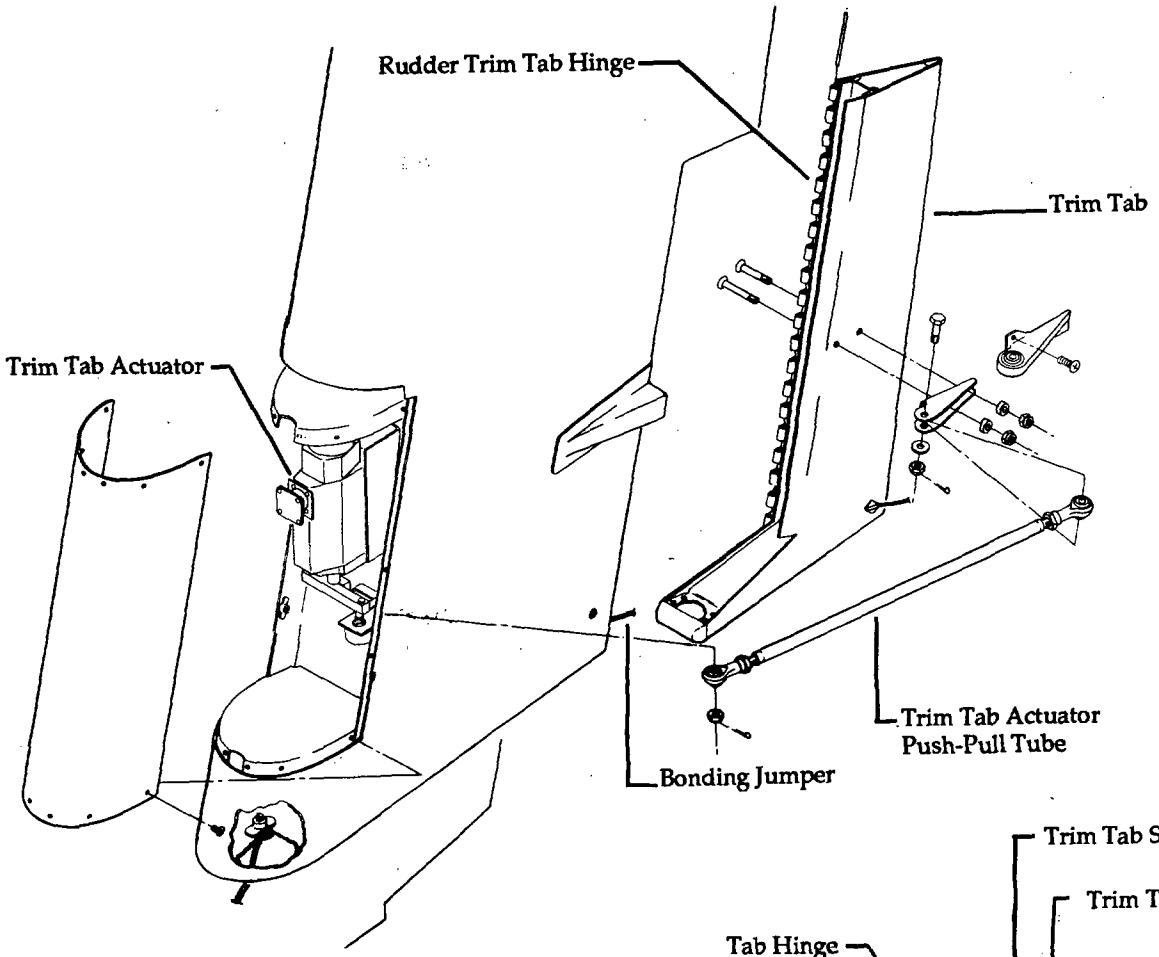
      NOTE:
      - Dial indicator must be calibrated within 12 months after being placed into service and every 12 months thereafter.
      - Equivalent substitutes may be used in lieu of the following:

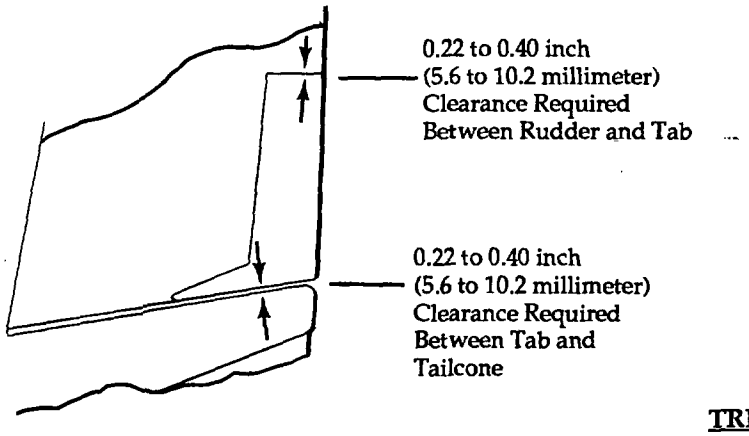
NAME	PART NUMBER	MANUFACTURER	USE
Aileron/Rudder Trim Tab Free Play Tool	2370120-1	Learjet Inc. Wichita, KS	Hold dial indicator
Aileron/Rudder Trim Tab Free Play Tool	2370120-2	Learjet Inc. Wichita, KS	Hold torque wrench
Torque wrench Click-type		Commercially Available	Apply pressure to trim tab

   B. Measure Rudder Trim Tab Free Play. (See Figure 202.)

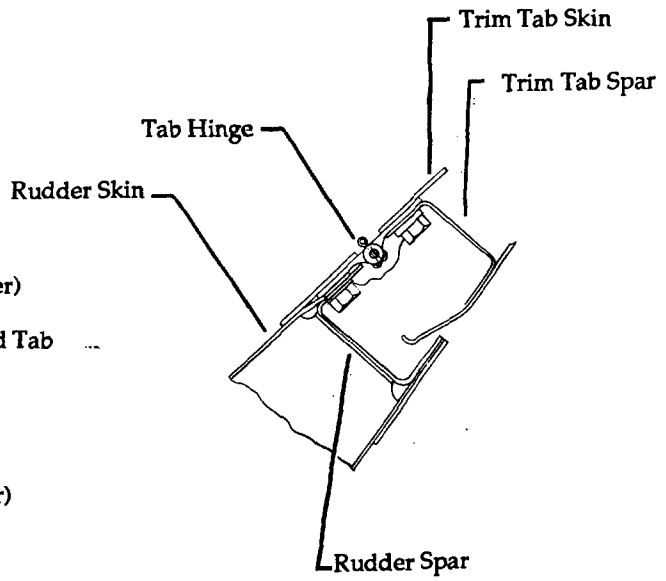
      NOTE: Perform rudder trim tab free play measurement in accordance with the current inspection intervals specified in Chapter 5.

      (1) Block rudder to prevent movement.
      (2) Streamline rudder trim tab.
      (3) Position -1 Aileron/Rudder Trim Tab Free Play Tool on rudder.
         (a) Adjust pads on clamp arms to apply adequate pressure on rudder to hold tool in place.





TRIM TAB CLEARANCES



TRIM TAB HINGE INSTALLATION DETAILS

Rudder Trim Tab Installation
Figure 201

5-2C
5-12B
5-9B

EFFECTIVITY: ALL

MM-99

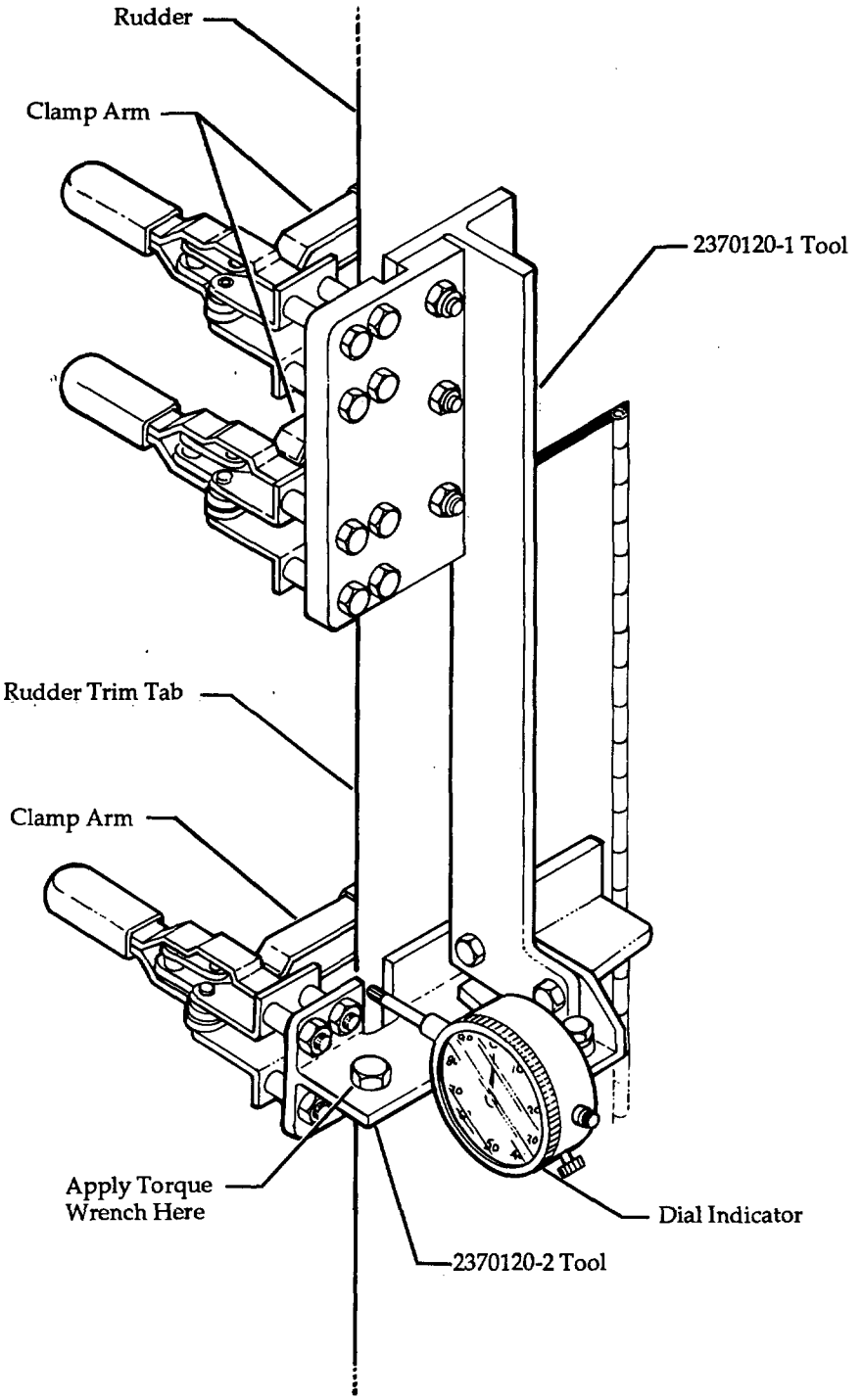
(4) Position -2 Aileron/Rudder Trim Tab Free Play Tool on rudder trim tab as close to dial indicator as possible. (See Figure 202.)

CAUTION: DO NOT APPLY ENOUGH FORCE TO PERMANENTLY DEFORM RUDDER OR TRIM TAB SURFACE.

(a) Adjust pads on clamp arm to apply sufficient clamping force to hold tool in place.

NOTE: Aileron/Rudder Trim Tab Free Play Tools shall not rock.

(5) Zero dial indicator.
(6) With slight finger pressure only, check for excessive free play.
  (a) If free play is more than 0.020 inch in either direction, check trim tab actuator linkage for proper adjustment.
  (b) Adjust or replace rudder trim tab actuator linkage components to eliminate free play. (Refer to 27-20-06, Removal/Installation.)
(7) Set torque wrench to 35 inch pounds and lock.
(8) Position torque wrench on 7/16 bolt head on top of -2 Aileron/Rudder Trim Tab Free Play Tool with handle perpendicular to control surface. (See Figure 202.)
(9) Verify that dial indicator is at zero and slowly apply left pressure to wrench. Watching dial indicator, note and record reading when torque wrench releases. (First reading.)
(10) Repeat step (9) three times to verify results, recording each reading. (See Figure 203.)
(11) Average closest two readings. (Example: 0.016, 0.019 and 0.017; eliminate 0.019 and add 0.016 and 0.017 and divide by 2.) (See Figure 203.)
(12) Repeat steps (7), (8), (9), (10) and (11) applying right pressure to torque wrench.
(13) Add measured free play in both directions to determine total. If total free play exceeds 0.040 inch, replace trim tab hinge pin and repeat inspection. If hinge pin has previously been replaced, replace trim tab hinge and repeat inspection.
(14) After free play measurement is complete remove Aileron/Rudder Trim Tab Free Play Tools from aircraft and unblock rudder.
(15) Return aircraft to normal.



(TYPICAL)

14-232B

Rudder Trim Tab Free Play Measurement
Figure 202

Learjet

RUDDER TRIM TAB FREE PLAY WORKSHEET

MEASURE LEFT FREE PLAY

DIAL INDICATOR READING NO. 1 _____

DIAL INDICATOR READING NO. 2 _____

DIAL INDICATOR READING NO. 3 _____

ADD CLOSEST TWO READINGS _____ + _____ = _____

DIVIDE TOTAL BY 2 _____ / 2 = _____ (LEFT FREE PLAY)

MEASURE RIGHT FREE PLAY

DIAL INDICATOR READING NO. 1 _____

DIAL INDICATOR READING NO. 2 _____

DIAL INDICATOR READING NO. 3 _____

ADD CLOSEST TWO READINGS _____ + _____ = _____

DIVIDE BY 2 _____ / 2 = _____ (RIGHT FREE PLAY)

TOTAL FREE PLAY

LEFT FREE PLAY _____

RIGHT FREE PLAY + _____

TOTAL FREE PLAY = _____

NOTE: Total Rudder Trim Tab free play shall not exceed 0.040 inch.

Rudder Trim Tab Free Play Worksheet
Figure 203

RUDDER PEDALS AND SUPPORT ASSEMBLY - MAINTENANCE PRACTICES

1. Removal/Installation

A. Removal of the Rudder Pedal and Support Assembly (See Figure 201.)

(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Rigging Pin (0.1875 inch [4.76 mm])	3170037-2	Learjet Inc. Wichita, KS	Hold the outboard bellcrank assembly in the rigged position.

(2) Remove the crew seats from the aircraft. (Refer to 25-10-01.)

(3) Remove floorboards and inner skin, as required, to gain access to rudder pedal support assemblies.

(4) Insert a rigging pin in each of the outboard bellcrank assemblies.

(5) Remove the attaching parts that attach the bobweight mechanical linkage to the copilot's side of the control column.

(6) Remove the attaching parts that attach the rudder pedal bellcrank push-pull tube to the lower pedal support assembly.

(7) Remove the attaching parts that attach the power brake link from the rudder pedal.

(8) When removing copilot's inboard pedal, remove the follow-up linkage from the rudder pedal. (Refer to 32-50-03.)

(9) Remove the attaching parts that attach the outboard end of rudder pedal support stud.

(10) Pull the support stud inboard (into the wheel well) until the outboard rudder pedal is released.

NOTE: If the nose gear steering actuator interferes with the removal of the support stud, place the aircraft on jacks and remove the nose gear strut. (Refer to 32-20-01.)

(11) Remove the rudder pedal from the aircraft.

(12) Continue to pull the support stud inboard (into the wheel well) until the inboard rudder pedal is released.

(13) Remove the rudder pedal from the aircraft.

(14) If a rudder pedal is to be remove from the pedal support, do the steps that follow:

(a) Remove the roll pin that attaches the pin to the pedal support.

(b) Pull the pin out of the pedal support and remove the rudder pedal from the support.

(c) Check the condition of the bushings. Replace them if necessary.

B. Installation of the Rudder Pedal and Support Assembly (See Figure 201.)
(1) If a rudder pedal is to be remove from the pedal support, do the steps that follow:
(a) If the bushings are replaced, lubricate them with (OGP) oil. (Refer to 12-20-00.)
(b) Set the bushings at their correct location in the rudder pedal.
(c) Position the rudder pedal on the support assembly.
(d) Install the pin that attaches the rudder pedal to the support assembly.
(e) Install the roll pin that attaches the pin to the support assembly.
(2) Make sure that the washer, nut, and cotter pin are installed on the inboard end of support stud.
(3) Align inboard rudder pedal support assembly bearings with pedal support stud hole at wheel well structure and insert support stud through bearings.
(4) Align outboard rudder pedal support assembly bearings with pedal support stud hole in structure and insert support stud through bearings and structure. Secure support stud outboard end with attaching hardware.
(5) Install the attaching parts that attach the rudder pedal bellcrank push-pull tube to the lower pedal assembly.
(6) Install the attaching parts that attach the power brake link to the rudder pedal assembly.
(7) Remove the rigging pins from the outboard bellcrank assemblies.
(8) Check the power brake valve rigging. (Refer to 32-43-02.)
(9) When the copilot's inboard pedal, install the follow-up linkage on the rudder pedal. (Refer to 32-50-03.)
(10) Install the attaching parts that attach the bob weight mechanical linkage to the control column.
(11) Install the nose gear strut, if previously removed. (Refer to 32-20-01.)
(12) Install and secure all floorboards and inner skins.
(13) Install the crew seats. (Refer to 25-10-01.)

2. Repairs

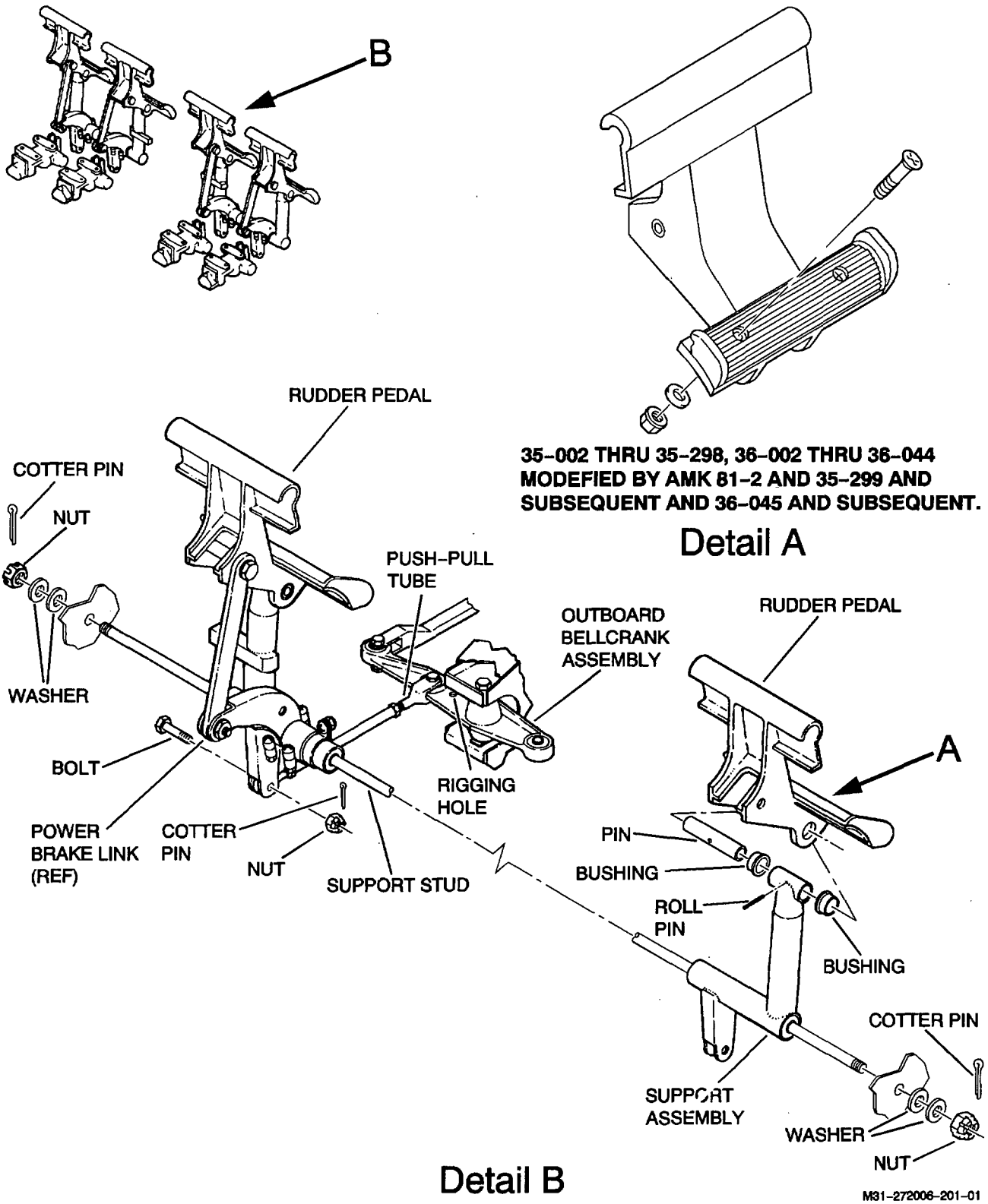
A. Replacement of the Anti-Slip Tread
(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Adhesive, Class I		Refer to 20-11-00.	Bonding tread to the rudder pedal.
Rubber Roller		Commercially Available	Install tread on rudder pedal.
Cleaning Solvent		Refer to 20-12-00.	Clean the rudder pedal.

(2) Remove all of the old anti-slip tread from the rudder pedal.
(3) Clean the old adhesive from rudder pedal using cleaning solvent. (Refer to 20-12-00.)
(4) Apply the adhesive to mating surface of rudder pedal.

(5) Allow adhesive to air dry to a tack condition (usually 5 to 10 minutes, depending on humidity and temperature). Test the adhesive by touching it with your finger. If adhesive is sticky to touch, but there is no transfer of adhesive to your finger, bond can be made.
(6) Remove the protective liner from anti-slip tread and apply anti-slip tread to rudder pedal.
(7) Using a rubber roller, firmly press anti-slip tread to the rudder pedal.
(8) Allow adhesive to air dry at room temperature for 24 hours.



Rudder Pedal and Support Installation
Figure 201

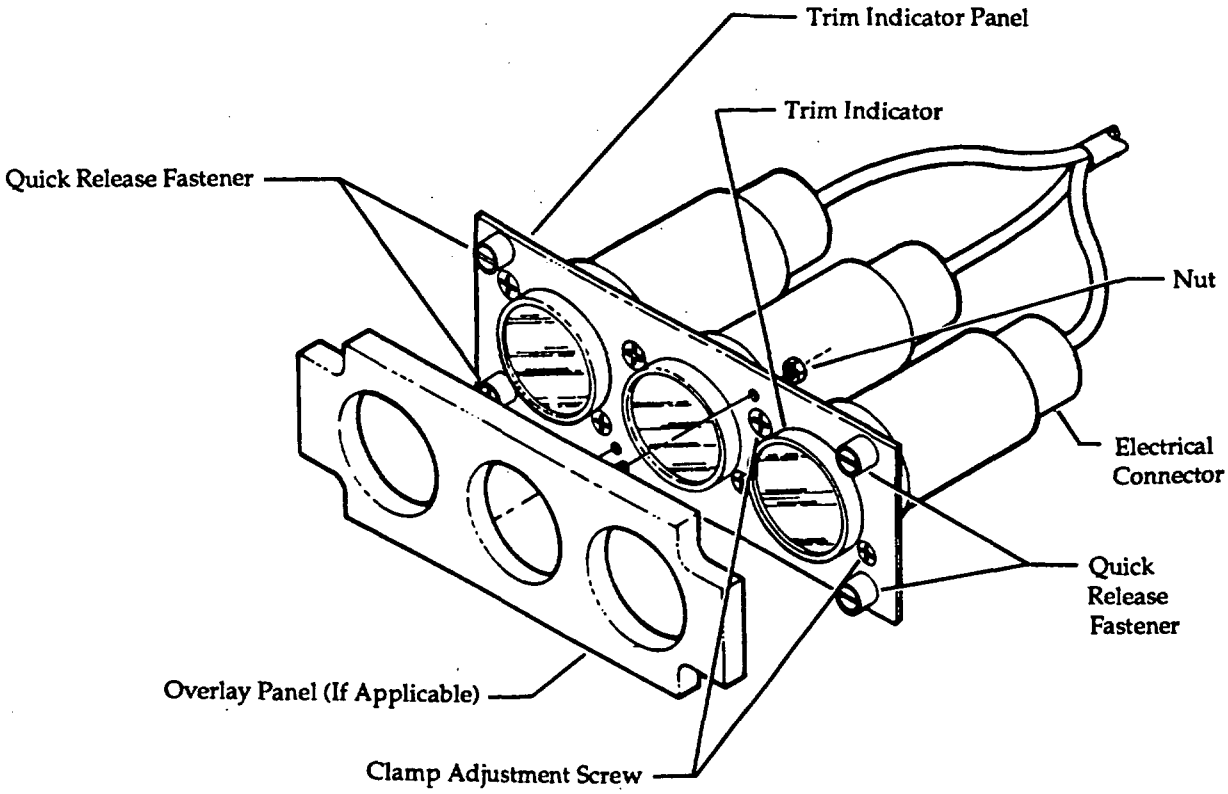
RUDDER TRIM POSITION INDICATOR - MAINTENANCE PRACTICES

1. **Removal/Installation** (See figure 201.)
   A. Remove Rudder Trim Indicator
      (1) Remove electrical power from aircraft.
      (2) Release trim panel assembly from the pedestal assembly by loosening quick release fasteners.
      (3) If applicable, remove parts securing overlay panel to face of trim panel and remove overlay.
      (4) Disconnect electrical connector from rudder trim indicator.
      (5) Loosen clamp adjustment screw and remove rudder trim indicator from trim indicator panel.
   B. Install Rudder Trim Indicator
      (1) Position rudder trim indicator in trim indicator panel and secure by tightening clamp adjustments screw.
      (2) Connect electrical connector to rudder trim indicator.
      (3) If applicable, position overlay panel on face of trim indicator panel and secure with attaching parts.
      (4) Position trim indicator panel in center pedestal and secure with quick release fasteners.
      (5) Restore electrical power to aircraft.
      (6) Perform Adjustment/Test of rudder trim indicator.

2. **Adjustment/Test** (See figure 202.)

   NOTE:
   - The rudder trim position indicator, located on the cockpit center pedestal, indicates rudder trim tab travel. Indicator input is from a potentiometer attached to the rudder trim tab actuator.
   - Calibration of the rudder trim indicator is performed at the position indicator calibration assembly located inside of the trim switch panel.

   A. The rudder trim tab potentiometer must be approximately centered with tab in neutral position prior to calibrating the rudder trim indicator.
   B. Calibrate rudder trim position indicator as follows:
      (1) Remove equipment as necessary from the pedestal adjacent to the trim switch panel. (Refer to Chapter 31.) This allows access to the position indicator calibration assembly.
      (2) Actuate rudder trim tab to neutral position.
      (3) Adjust Yaw Zero potentiometer to obtain zero indication on the rudder trim indicator.
      (4) Actuate rudder trim tab to full left.
      (5) Adjust Yaw Expand potentiometer to obtain full left indication on the rudder trim indicator.
      (6) Actuate rudder trim tab to full right.
      (7) Adjust Yaw Expand to obtain full right indication on the rudder trim indicator.
      (8) Cycle rudder trim tab while checking respective indications on the rudder trim indicator.
      (9) Readjust Yaw Expand potentiometer as necessary to obtain coordinated indications.



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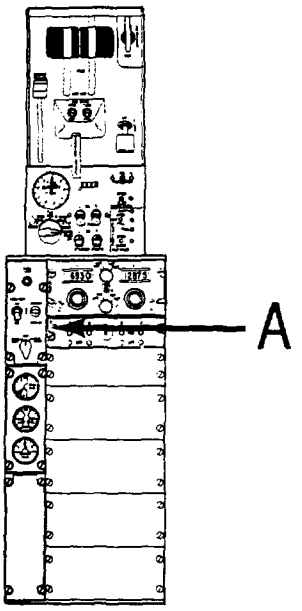
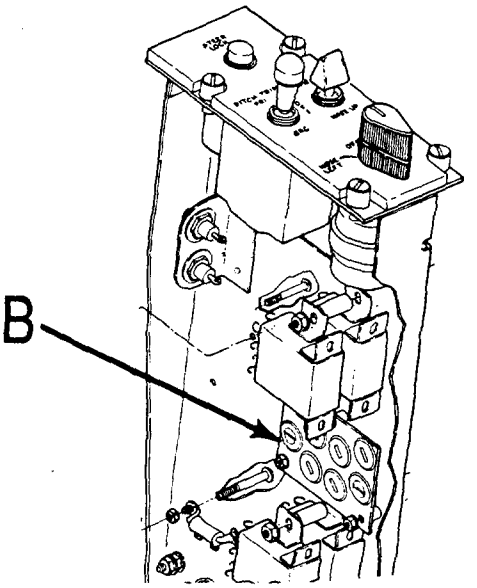
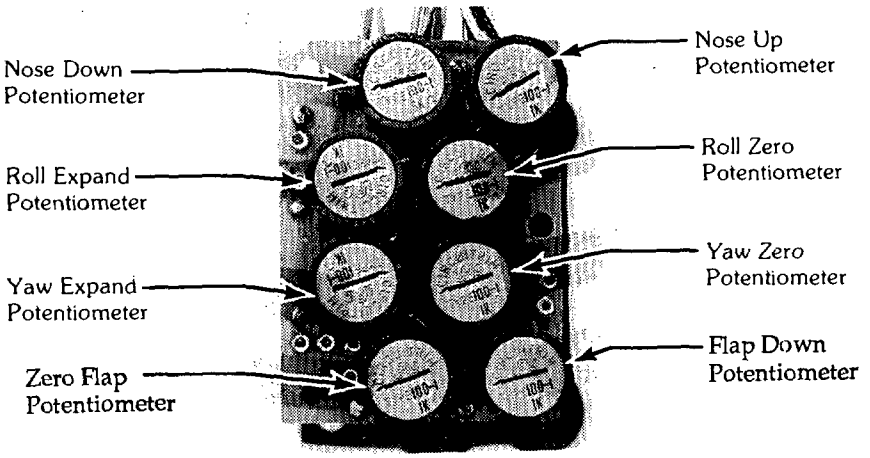
Rudder Trim Indicator Installation
Figure 201



(TYPICAL)



Detail A



Detail B

Position Indicator Calibration Assembly
Figure 202

TRIM TAB ACTUATOR - MAINTENANCE PRACTICES

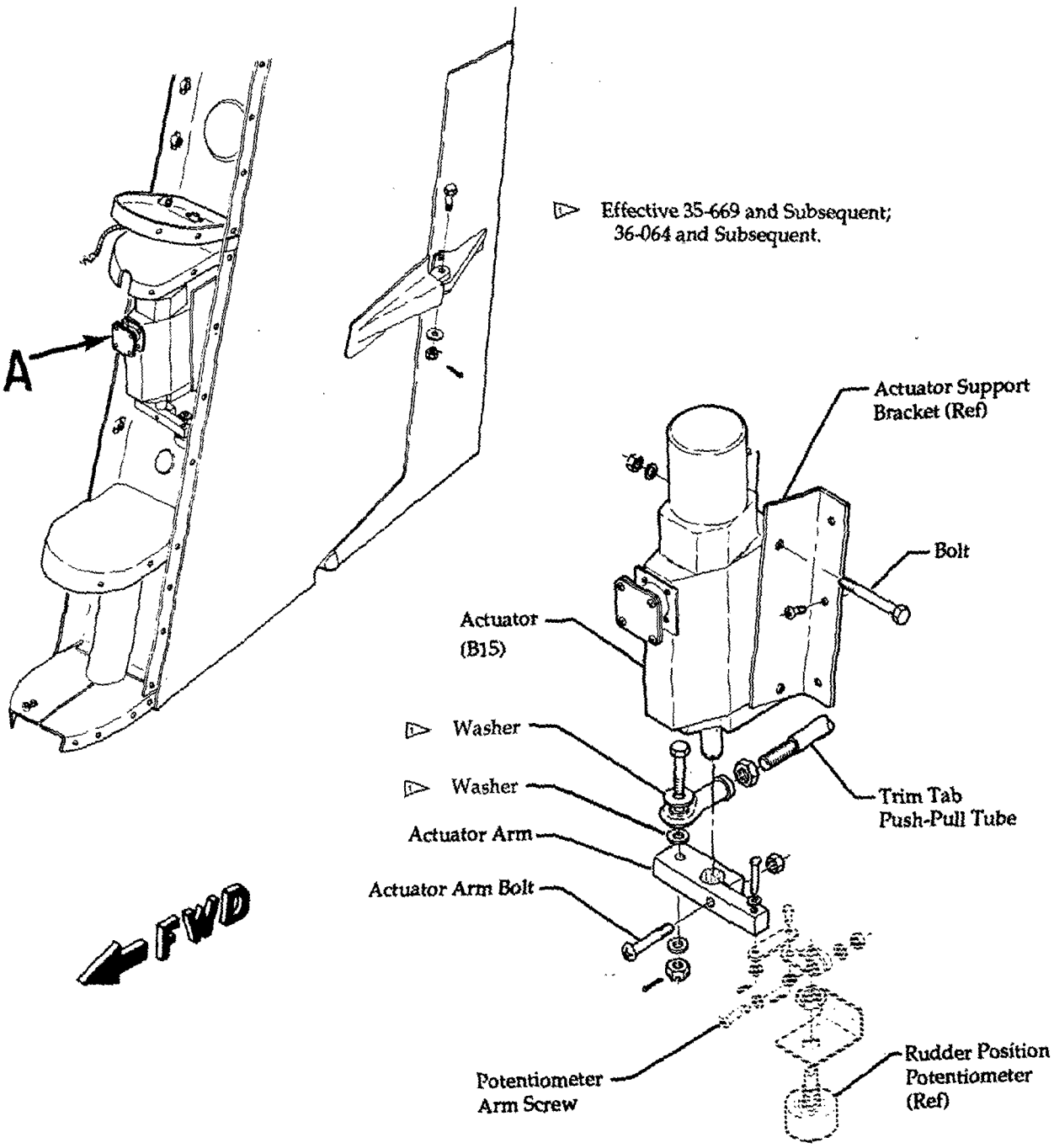
1. Removal/Installation

A. Remove Rudder Trim Tab Actuator (See figure 201.)

(1) Remove lower access cover from rudder leading edge.
(2) Remove rudder bellcrank access cover from top of tailcone.
(3) Disconnect push-pull tube from tab actuator arm.
(4) Loosen potentiometer arm screw at potentiometer shaft and remove arm from shaft.
(5) locate actuator electrical connector in tailcone, disconnect connector, and cut electrical wire (approximately 6 inches (15.2 centimeters) from connector) to allow for splicing later.
(6) Tie a string approximately 4 feet (122 centimeters) long to cut end of electrical wires.
(7) Remove attaching parts and actuator (with trim tab push-pull tube, actuator arm, link, and potentiometer arm attached) from the rudder.
(8) Pull electrical wiring free from rudder torque tube. Remove string from electrical wiring. Leave string in torque tube for reinstalling actuator.
(9) Loosen actuator-arm bolt and remove actuator arm from actuator shaft.

B. Install Rudder Trim Tab Actuator (See figure 201.)

(1) Position actuator arm on actuator shaft.
(2) Align actuator arm index spot with actuator shaft index line. Tighten actuator arm bolt.
(3) Position actuator on actuator support brackets and secure with attaching parts.
(4) Connect push-pull tube to trim tab.
(5) Align potentiometer arm on potentiometer shaft. Tighten screw.
(6) Tie string to actuator electrical wiring. Pull electrical wiring through torque tube. Splice electrical wiring to electrical-plug wiring. Connect plug.
(7) Check rudder trim tab for proper operation. (Refer to 27-20-00.)
(8) Install lower access cover on rudder leading edge.
(9) Install rudder bellcrank access cover on top of tailcone.



Detail A

Rudder Trim Tab Actuator Installation
Figure 201

RUDDER TRIM TAB ACTUATOR POTENTIOMETER - MAINTENANCE PRACTICES

1. Tools and Equipment

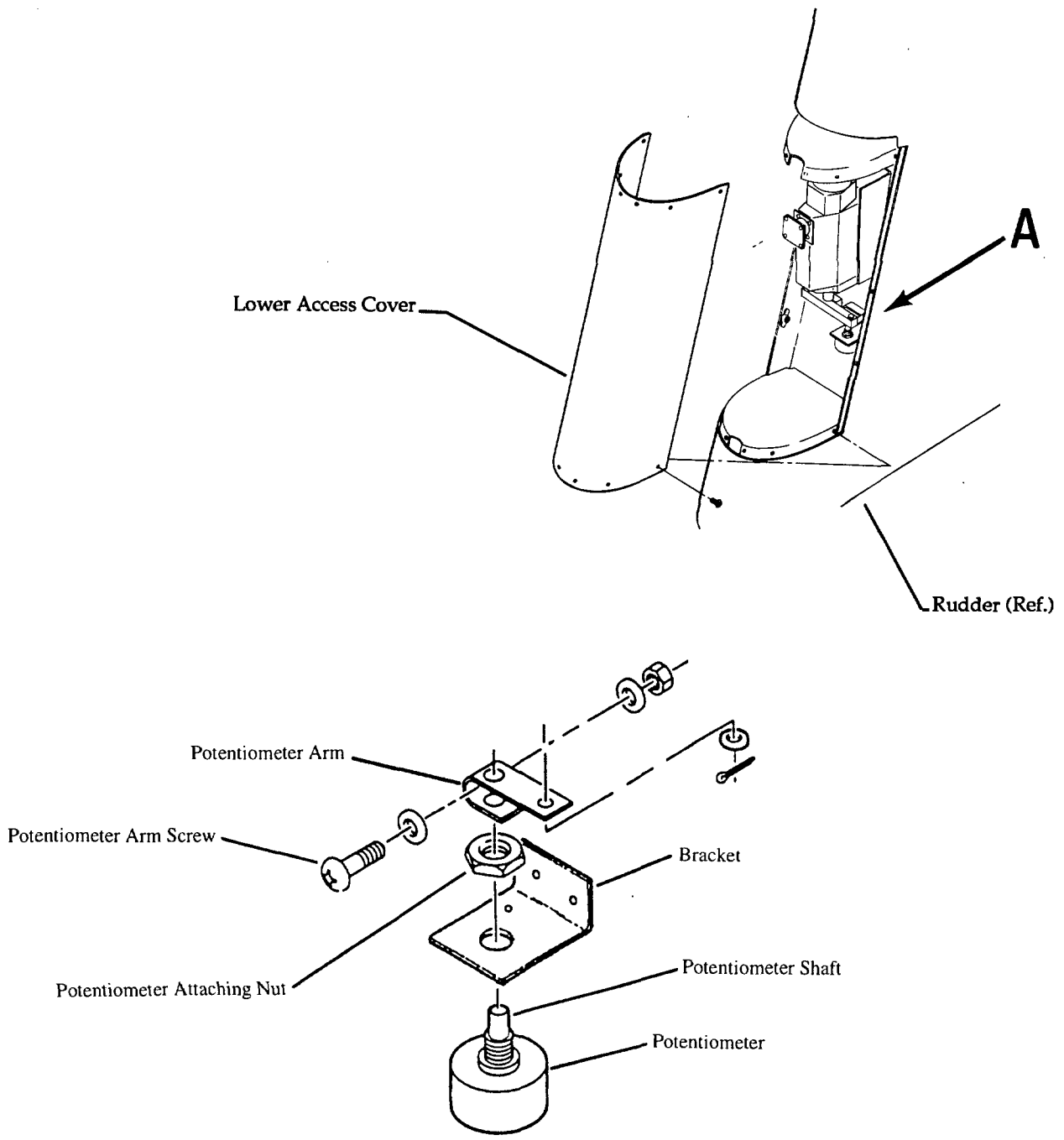
NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Multimeter	Model 260-7	Simpson	Resistance measurement.
Masking Tape		Commercially available	Secure trim tab.

2. Removal/Installation

A. Removal of Rudder Trim Tab Actuator Potentiometer (See Figure 201.)
   (1) Remove lower access cover from rudder leading edge.
   (2) Loosen potentiometer arm screw at potentiometer shaft. Remove arm from shaft.
   (3) Remove potentiometer attaching nut and potentiometer from bracket.
   (4) Disconnect electrical wiring at potentiometer.
   (5) Remove potentiometer from rudder.

B. Installation of Rudder Trim Tab Actuator Potentiometer (See Figure 201.)
   (1) Connect potentiometer electrical wiring, in rudder, to potentiometer.
   (2) Use existing hardware to install potentiometer on bracket.
   (3) Tape trim tab to neutral position.
   (4) Use an ohmmeter to measure potentiometer resistance. Rotate potentiometer shaft to center position (± 10 ohms). Disconnect ohmmeter
   (5) Being careful not to move potentiometer shaft; position potentiometer arm on potentiometer shaft, and tighten potentiometer arm screw.
   (6) Check rudder trim tab for proper operation. (Refer to 27-20-00.)
   (7) Install lower access cover.



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Rudder Trim Tab Actuator Potentiometer Installation
Figure 201

1. Removal/Installation

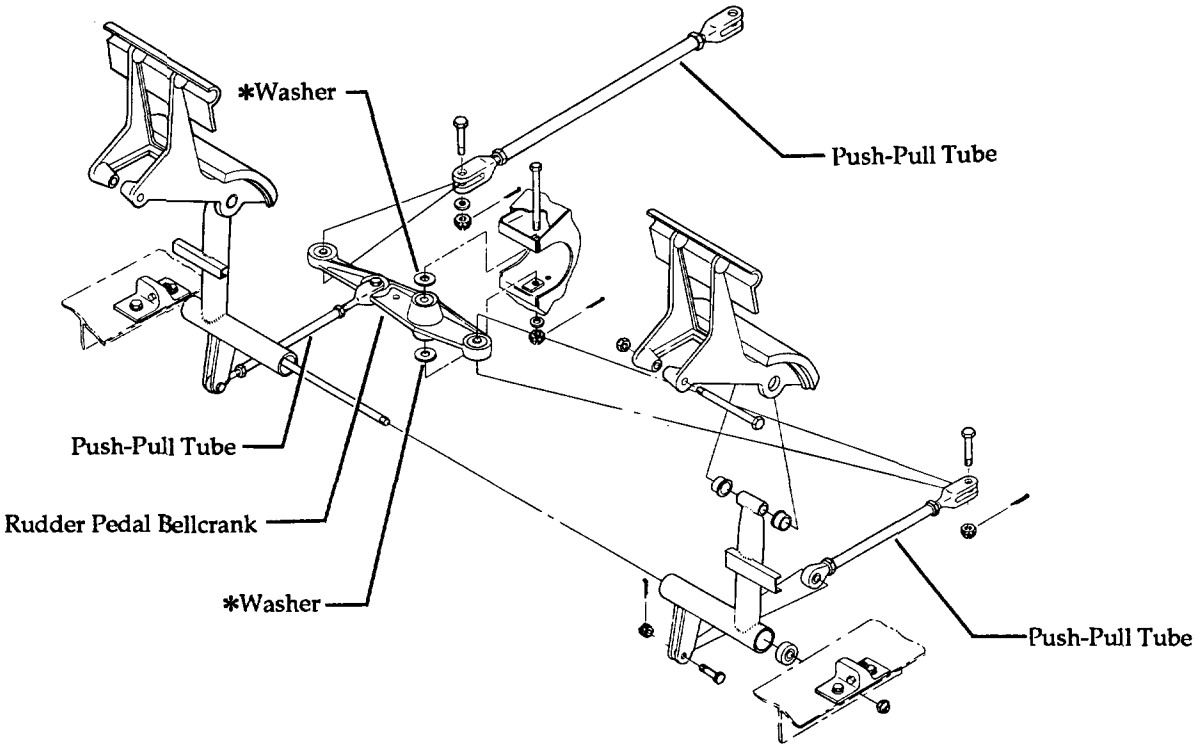
NOTE: Removal/Installation procedures are typical for both right and left Rudder Control Bellcrank Assemblies.

A. Remove Rudder Control Bellcrank Assembly (See figure 201.)
   (1) Remove electrical power from aircraft.
   (2) Remove applicable cockpit seating, carpeting, and floorboards as required, to gain access to rudder bellcrank assemblies.
   (3) Remove attaching parts securing rudder control pushrod assemblies to rudder control bellcrank.
   (5) Remove attaching parts securing rudder control bellcrank assembly to bellcrank support bracket; remove bellcrank assembly from aircraft.

B. Install Rudder Control Bellcrank Assembly (See figure 201.)

CAUTION: **WHEN REPLACING RUDDER CONTROL BELLCRANK, SHIM AS REQUIRED USING AN960KD WASHERS. ENSURE MAXIMUM GAP DOES NOT EXCEED 0.010 INCH.**

   (1) Position rudder control bellcrank assembly at its appropriate location in bellcrank support brackets. Secure with attaching parts.
   (2) Position rudder control push-pull rods and push-pull tubes at their appropriate locations up to rudder pedal bellcrank, and secure with attaching parts.
   (3) Operationally check rudder control system. (Refer to Inspection/Check, 27-20-00.)
   (4) Check rudder control system rigging. (Refer to 27-20-00.)
   (5) Install all previously removed floorboards, carpeting, and cockpit seating.
   (6) Restore aircraft to normal.



* When replacing rudder control bellcrank, shim as required using AN960KD washers. Ensure maximum gap does not exceed 0.010 inch.

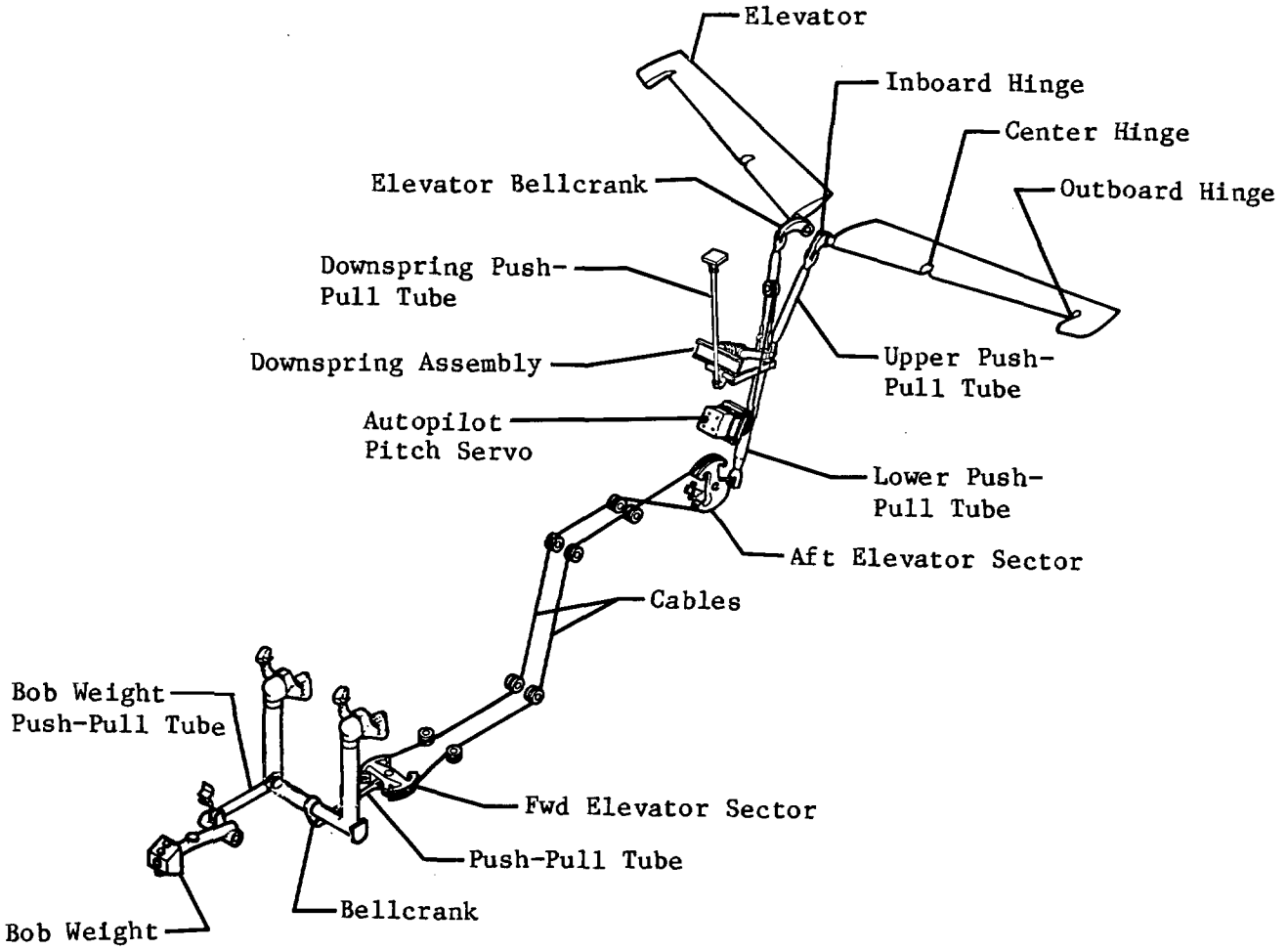
Rudder Control Bellcrank Installation
Figure 201

ELEVATOR - DESCRIPTION AND OPERATION

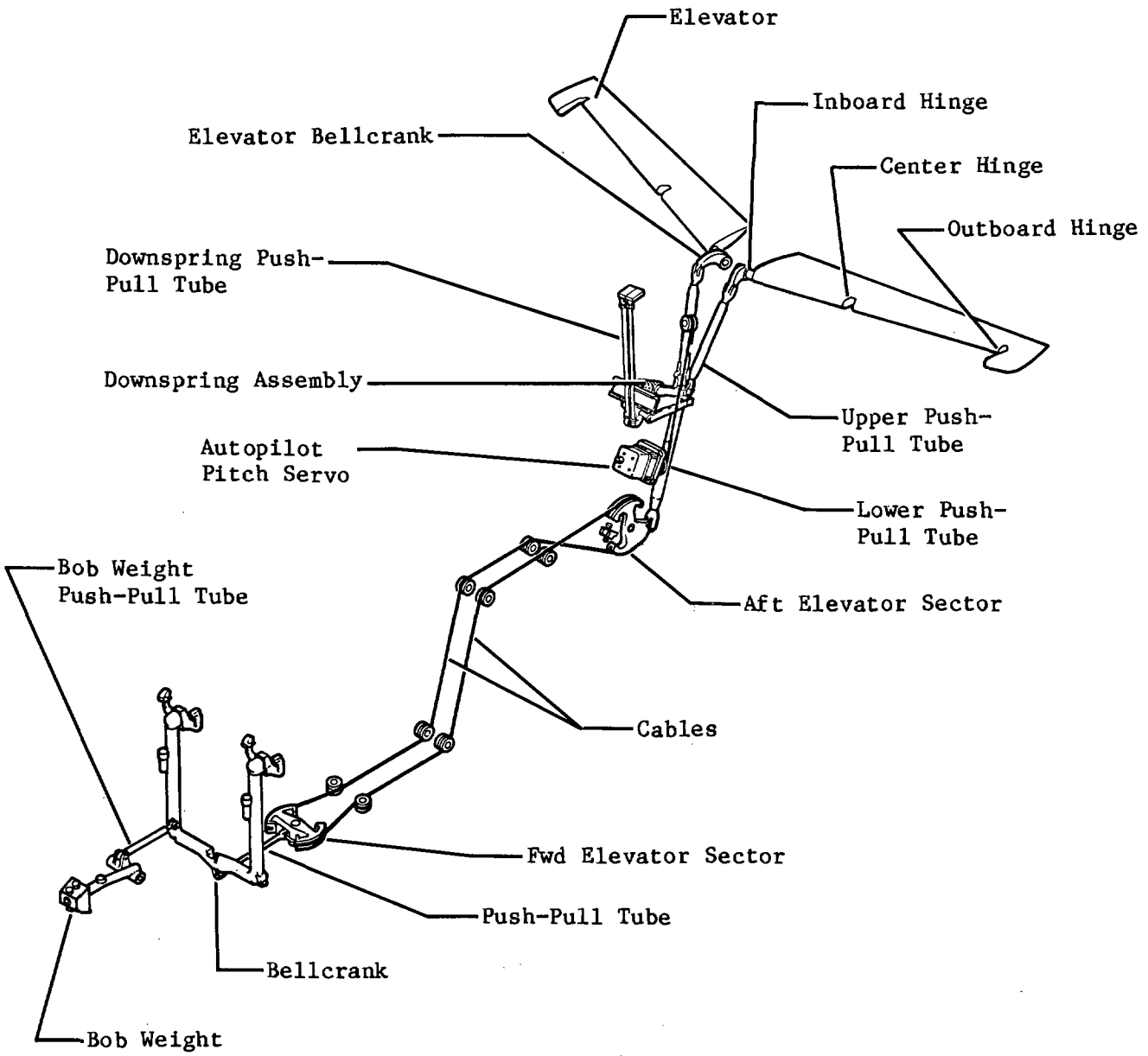
1. DESCRIPTION

A. The elevators provide longitudinal control of the aircraft. Each elevator is connected to the horizontal stabilizer at three hinge points. The mechanical control system consists of control column bellcranks, sectors, cables, and push-pull tubes. Mounted on the control column is a bellcrank which, through a push-pull tube, provides rotary motion to the forward elevator sector. The forward elevator sector is connected to the aft elevator sector by a series of cables and pulleys. The aft sector operates the elevators through bellcranks, push-pull tubes, and a downspring assembly. Access to the elevator control cables and push-pull tubes is provided by access panels in the floorboard lower fuselage, aft tailcone, and vertical stabilizer. The elevators are equipped with a fixed (ground adjustable) tab on the trailing edge. The fixed tab provides better control characteristics of the elevators and is set in the 9° up position at the factory.

B. Elevator control motion is transferred from the interconnected control columns, through a push-pull tube, bellcrank assembly, and closed loop cable assembly to a sector assembly in the tailcone. From the sector assembly, motion is transferred through push-pull tubes to a sector on the elevators. A bob weight, linked to the right control column, extends forward through frame 5 into the nose compartment to provide an adjustable static balance device in the elevator control system. To decrease horizontal stabilizer pitch trim force, a downspring installation is provided. The down spring interconnects the horizontal stabilizer and elevator in such a manner that movement of the stabilizer creates tension on the downspring which causes a proportional movement of the elevator in the opposite direction. The autopilot servo is connected to the yoke assembly by servo cables. The autopilot capstan slip clutch is set to allow the pilot override of the autopilot system. (Refer to 22-13-00.)



Elevator Control System Installation
Figure 1 (Sheet 1 of 2)



Elevator Control System Installation
Figure 1 (Sheet 2 of 2)

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	Measure angles.
Digital Protractor (DP-45)	02475-01	Lucas Schaevitz Pennsauken, NJ	Measure angles.
Contour Plate	2471002	Learjet Inc. Wichita, KS	Obtain elevator neutral position.
Contour Board	2323000	Learjet Inc. Wichita, KS	Check horizontal stabilizer travel.
Caging Pin		Local Manufacture	Cage downspring.
Spring Scale - Push-Pull (Accurate to 1/2 pound)		Commercially Available	Measure control column force.
Rigging Pin	3170037	Learjet Inc. Wichita, KS	Elevator rigging.
Tensiometer Adapter*	2471007	Learjet Inc. Wichita, KS	Measure cable tension.
Tensiometer	T60-1001-C8-1A	Learjet Inc. Wichita, KS	Measure cable tension.

* The tensiometer adapter is intended for use on servo cables in areas that are inaccessible to normal tensiometer usage.

2. Adjustment/Test

A. Rig Elevator Control System (See Figure 201.)

(1) Remove floorboards and external access covers as required to gain access to forward and aft elevator sectors.

NOTE:

- Rigging pin is fabricated from 3/16 inch (4.76 mm) round steel stock. Make 90° bend, approximately 8-1/2 inches (21 cm) from end of 12 inch (30.5 cm) piece of stock. Paint short end of piece red (for visibility).
- Caging pin is fabricated from 3/16 inch (4.76 mm) round steel stock. Make 90° bend, approximately 3 inches (7.35 cm) from end of 6 inch (15.3 cm) piece of stock. Paint one end of piece red (for visibility).

(2) Set forward and aft elevator sectors in neutral and insert rigging pins through sectors.
(3) Level the aircraft. (Refer to Chapter 8.)
(4) On *Aircraft 35-002 thru 35-020 and 36-002 thru 36-013*, adjust push-pull tube between the forward elevator sector and control column bellcrank to set control column in neutral (3 [±1]° aft of vertical). On *Aircraft 35-021 and Subsequent and 36-014 and Subsequent*, adjust push-pull tube between the forward elevator sector and control column bellcrank to set control column in neutral (5 [±1/2]° aft of vertical).
(5) Position horizontal stabilizer to 1° 40' down from neutral (full aircraft nose down position.) (Refer to 27-40-00, Adjustment/Test.)
(6) Install contour board on horizontal stabilizer at RBL or LBL 19.40.
(7) Adjust LH elevator upper pushrod assembly until elevator contacts contour board. When elevator contacts contour board, then elevator is faired-in with horizontal stabilizer. Repeat steps (6) and (7) for RH side using the RH elevator upper pushrod.

NOTE: In the event that the contour board makes contact with the elevator fixed trailing tab (when installed), it is permissible to modify the plate to prevent any contact between the board and tab. (See Figure 201, Detail C for modification information.)

(8) Remove contour board.

WARNING: DO NOT BEND ELEVATOR ASSEMBLY TRAILING TAB TO CORRECT ANGULAR POSITION. IF TRAILING TAB ANGULAR POSITION IS INCORRECT, REPLACE WITH A NEW TAB. (REFER TO ELEVATOR INSTALLATION, 27-30-01.)

(9) On *Aircraft with elevator fixed trailing tabs installed*, use an adjustable protractor to check angular position of both LH and RH trailing tabs. On *Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not modified per AAK 76-4, "Reduced Approach Speed System"*, angular position shall be 12 (±2)° up. On *Aircraft 35-067 and Subsequent, 36-018 and Subsequent, and prior aircraft modified per AAK 76-4, "Reduced Approach Speed System"*, angular position shall be 9 (±2)° up.
(10) Install clinometer on LH or RH horizontal stabilizer. Adjust to zero.
(11) Adjust elevator control cable tension in accordance with procedures outlined in step 2.C.
(12) Adjust lower push-pull tube further to obtain 0° 24' trailing edge up from streamline neutral elevator position.
(13) With the elevators in the rig position (obtained in previous steps) adjust pitch servo cable tension to 60 (±5) pounds (27.2 [±2.27] kg) at 75°F (24°C). (Refer to 27-00-00 for cable tension temperature-load correction.)

NOTE: When cable turnbuckles are loosened or tightened to adjust cable tension, the elevator may come out of adjustment. It may be necessary to readjust the elevator to the neutral position. Repeat these procedures alternately until the elevator is in the rigged neutral position and at the required cable tension.

(14) Safety wire cable-adjustment turnbuckles.
(15) With control column in neutral, adjust bobweight push-pull tube so bobweight clears aircraft structure equally in both the up and down positions.
(16) Remove rigging pins.
(17) With the horizontal stabilizer at the 1°40' nose-down position, set the primary up stop bolts (located at aft sector in tailcone) to obtain elevator trailing edge up travel of 16° (+0', -30') from the rig position. On *Aircraft 35-040 and Subsequent and 36-017 and Subsequent and aircraft modified by SB 35/36-27-3, "Replacement of Primary Control System Stop Bolts,"* safety wire stop bolt.

(18) With the horizontal stabilizer at the 1°40' nose-down position, set the primary down stop bolt (located at aft sector in tailcone) to obtain elevator trailing edge down travel of 15° (±30') from the rig position. On *Aircraft 35-040 and Subsequent and 36-017 and Subsequent and aircraft modified by SB 35/36-27-3, "Replacement of Primary Control System Stop Bolts,"* safety wire stop bolt.

(19) Adjust secondary stop bolts on the forward elevator sector to contact their stops when 100 (±25) pounds (45.36 [±11.34] kg) of force is applied to the control column wheel (fore and aft) after primary stop bolts are contacted.

(20) Move control column forward and aft until secondary stops are contacted. Note the position of the bobweight with respect to aircraft structure at each extreme position. Adjust bobweight push-pull tube as required to prevent bobweight from contacting structure.

(21) Adjust bobweight stop bolts so that the secondary stop bolts on the forward sector and the bobweight stop bolts contact their stop simultaneously in either the up or down position.

(22) Move elevators until down-spring caging pin hole is exposed.

WARNING: THE EXISTING SPRING TENSION OF THE CAGED DOWN-SPRING ASSEMBLY IS APPROXIMATELY 300 POUNDS (136 KG). USE EXTREME CARE WHEN REMOVING ATTACHING PARTS AND SPRING ASSEMBLY FROM AIRCRAFT.

(23) Insert down-spring caging pin and remove bolt and nut attaching draw-bolt to frame assembly. (Refer to 27-30-01, Removal/Installation.)

(24) Position horizontal stabilizer to 5° 30' leading edge down from neutral. (Refer to 27-40-00, Adjustment/Test.)

(25) Remove screw and nut from LH side of vertical stabilizer (spar cap).

(26) Insert rigging pin through spar cap hole and into LH yoke assembly bracket.

(27) Remove safety wire, loosen jamnut(s), and adjust down-spring push-pull tube(s) until rigging pin can be inserted through the frame assembly and into the right bracket as shown.

(28) Tighten down-spring push-pull tube jamnut(s) and safety wire. Remove rigging pin and install screw and nut in spar hole. Reattach drawbolt to frame assembly and remove caging pin. (Refer to 27-30-01, Removal/Installation.)

(29) Operate elevators through several cycles while checking for freedom of movement.

(30) Perform elevator control system friction test as follows (with autopilot disengaged). (See Figure 202.):

(a) Position horizontal stabilizer full nose up (aircraft nose down) position.

(b) Place a push-pull spring scale (accurate within 1/2 pound [0.23 kg]) on the pilot's control wheel hand grip so that the force applied is through the horizontal centerline of the control wheel hub.

(c) Slowly pull horizontally (aft) and record the force applied to move the control column off of the forward stop. Enter this value in Figure 202.

(d) Continue pulling aft and record the force applied as the control column moves through the neutral position. Enter this value in Figure 202.

(e) Continue to pull the control column aft and record the force applied just before reaching the aft stop. Enter this value in Figure 202.

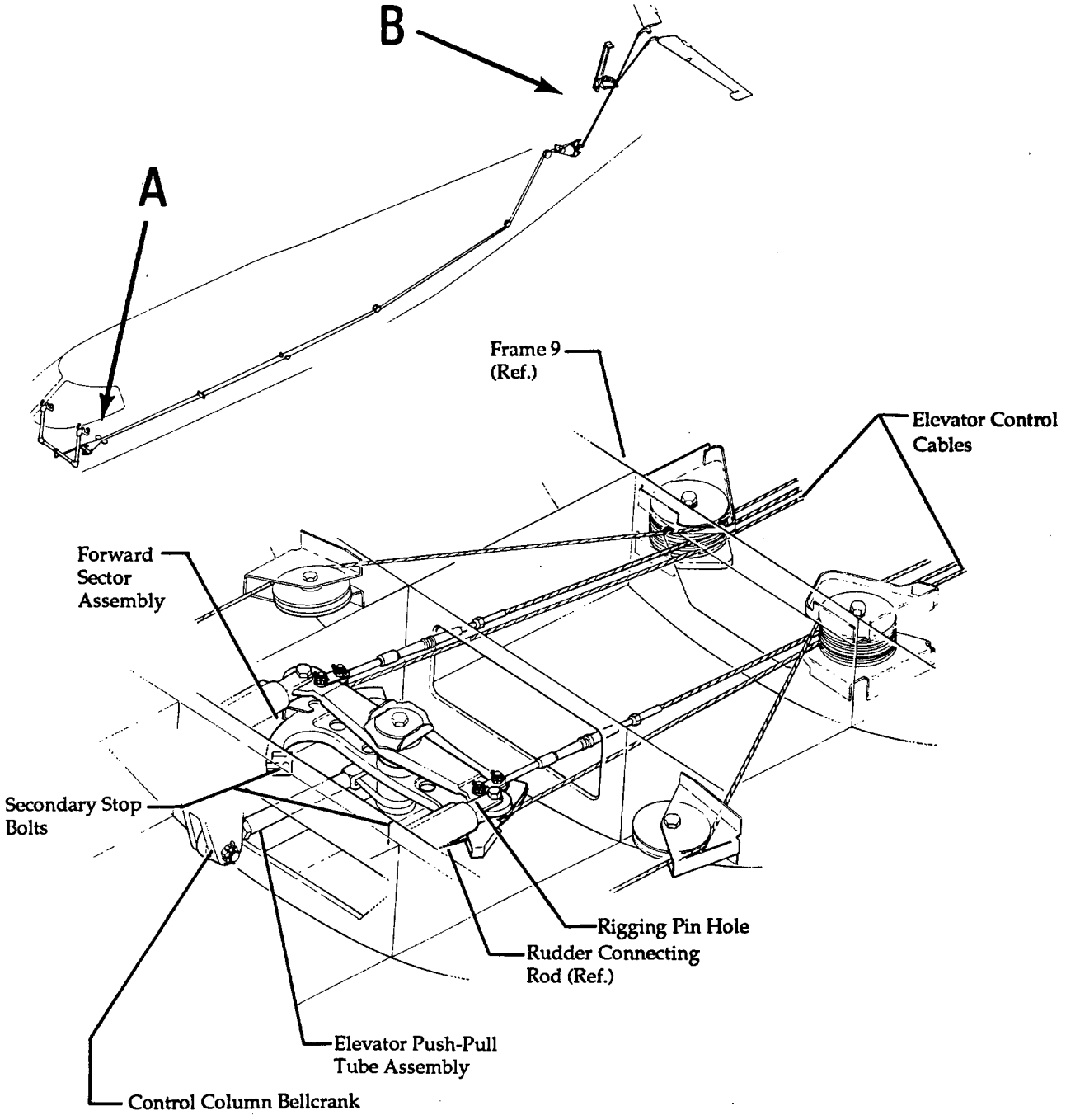
(f) With the control column held against the aft stop, slowly release the control column and record the release force just off of the aft stop. Enter this value in Figure 202.

(g) Continue to release the control column and record the release force as it travels through the neutral position. Enter this value in Figure 202.

(h) Continue to release the control column and record the release force just before it contacts the forward stop. Enter this value in Figure 202.

(i) Compute breakout force for each position (forward, neutral, and aft) by subtracting the release force from the pull force. enter these values in figure 202.

(j) Compute the average breakout force by adding the breakout force column forward, neutral, and column aft and divide that sum by 3. The breakout force average shall be less than 12 pounds (5.44 kg).

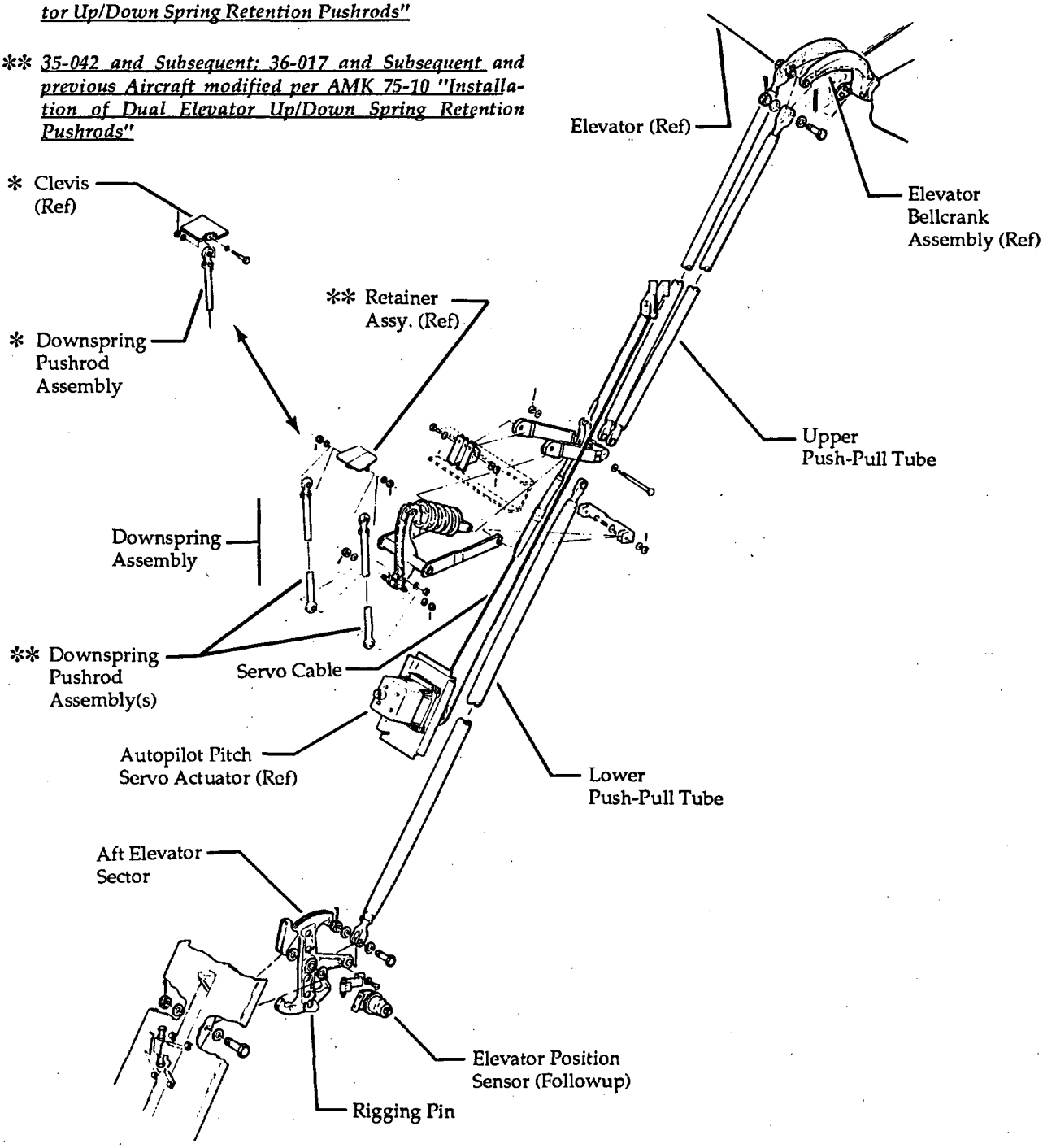


Detail A

Elevator Control System Rigging
Figure 201 (Sheet 1 of 4)

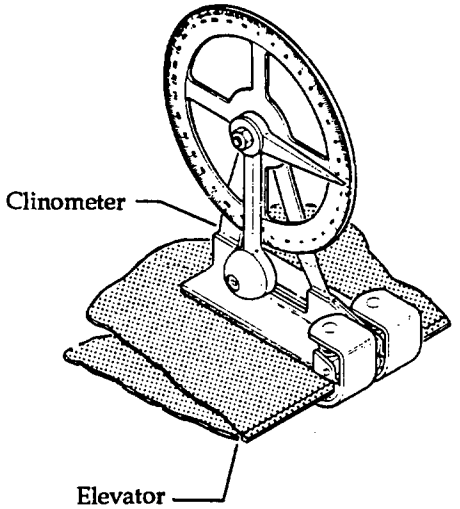
* *35-002 thru 35-041 and 36-002 thru 36-016 except when modified per AMK 75-10 "Installation of Dual Elevator Up/Down Spring Retention Pushrods"*

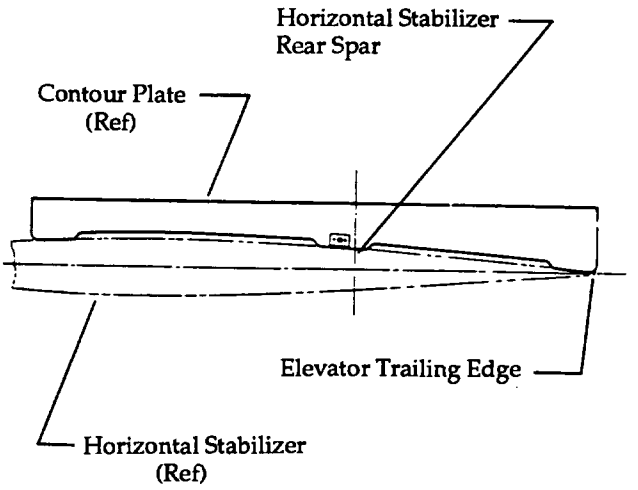
** *35-042 and Subsequent; 36-017 and Subsequent and previous Aircraft modified per AMK 75-10 "Installation of Dual Elevator Up/Down Spring Retention Pushrods"*

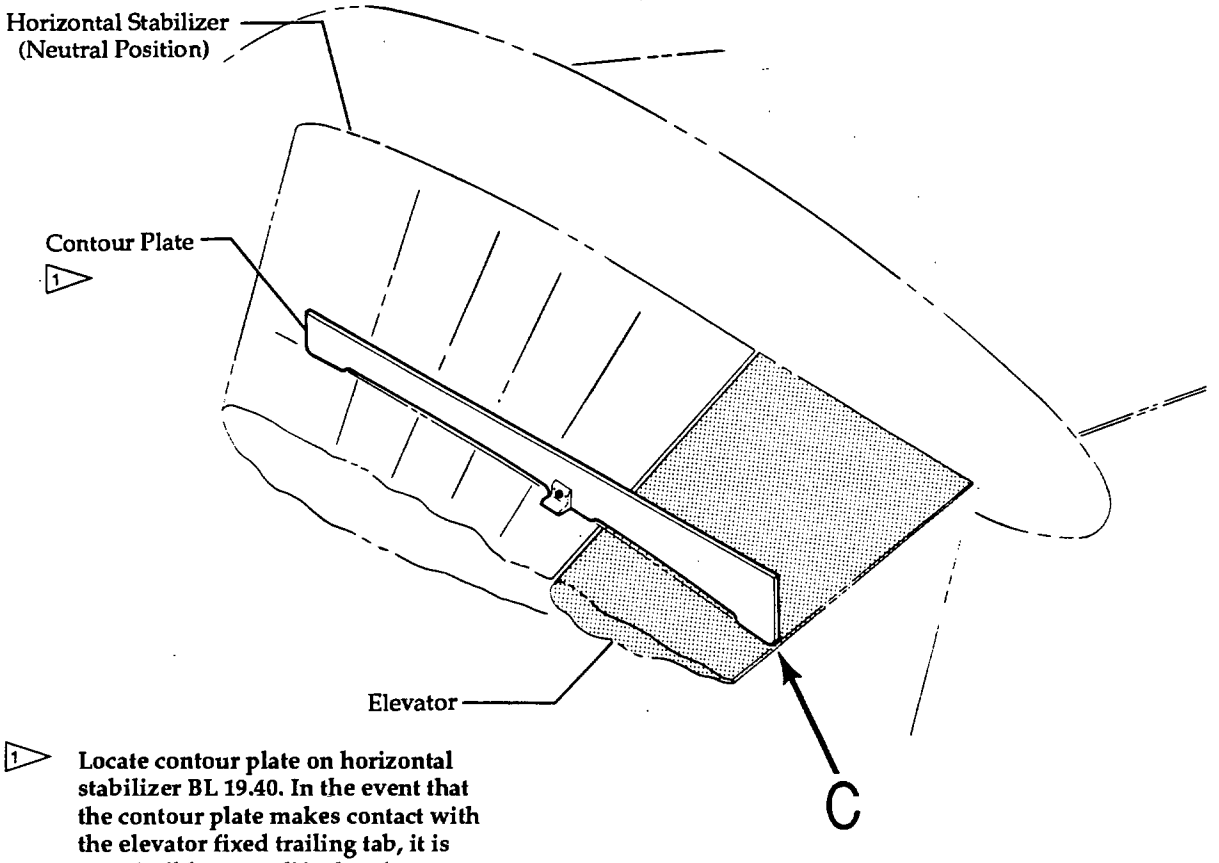


Detail B

Elevator Control System Rigging
Figure 201 (Sheet 2 of 4)







1 **Locate contour plate on horizontal stabilizer BL 19.40. In the event that the contour plate makes contact with the elevator fixed trailing tab, it is permissible to modify the plate, so as to prevent any contact between the plate and tab. (See Detail C.)**

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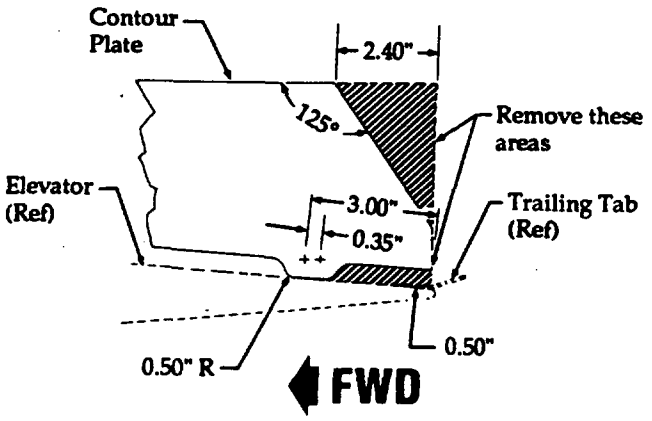
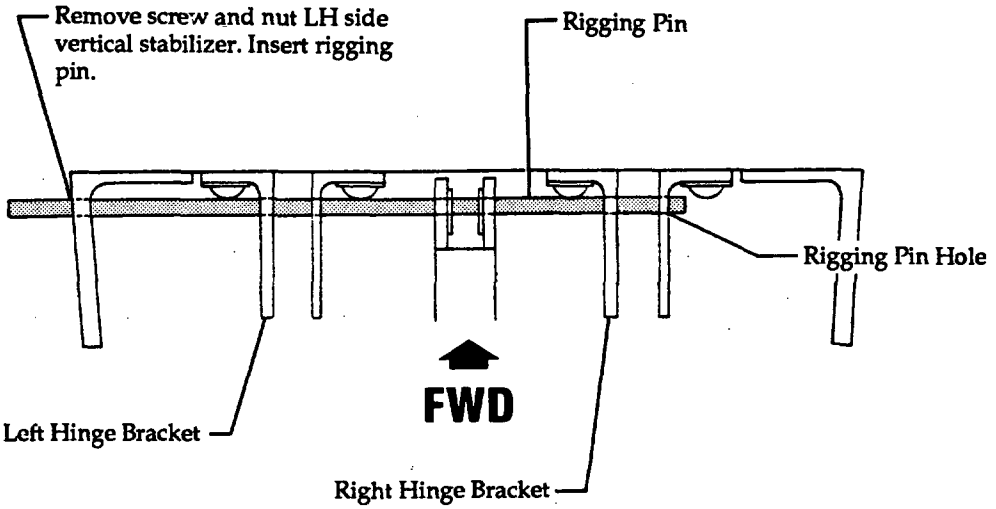
15-2C-1

Elevator Control System Rigging
Figure 201 (Sheet 3 of 4)



(CONTOUR PLATE MODIFICATION)

Detail C



(k) If the breakout force average is greater than 12 pounds (5.44 kg), check the following:
1) Proper cable tensions.
2) Pulleys, sectors, and bellcranks for freedom of movement; inspect system for binding.
3) System for proper lubrication.
4) Ensure that elevator control system is properly rigged.

(l) After all discrepancies have been corrected, perform steps (a) thru (j) and ensure that the breakout force average is less than 12 pounds (5.44 kg).

(31) Install and secure floorboards and access covers. Remove clinometer from elevator.

C. Elevator Control Cable Tension Adjustment (See Figure 203.)

NOTE:
- Prior to elevator control cable tension adjustment, torque wrench tensiometer adapter calibration must be accomplished. (Refer to 27-00-00, Adjustment/Test.)
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The cable tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to 27-00-00, Adjustment/Test.)

(1) Remove access cover located on LH side of tailcone allowing access to elevator aft sector assembly. With elevator in faired position, install rigging pin in elevator aft sector assembly.
(2) Remove forward keelbeam access panel to gain access to elevator control cables and turnbuckles.
(3) Remove locking clips from turnbuckles.
(4) If a new cable is being installed, perform the following:
(a) Adjust elevator cable tension to 200 (±10) pounds (90.7 [±4.5] kg).
(b) Cycle elevator system 25 times to seat in and pre-stretch new cable.
(c) Reduce cable tension.
(5) Adjust elevator control cable tension to 75 (±5) pounds at 75°F (34 [±2.3] kg at 24°C). (Refer to 27-00-00 for cable tension temperature-load correction table.)

NOTE:
- The threads at opposite ends of the turnbuckle barrel shall be started at the same time so thread engagement on both ends will be approximately equal.
- On *Aircraft with lead-acid batteries*, apply corrosion protection to control cable in area of aircraft batteries. (Refer to 27-00-00, Cleaning/Painting.)
- The final adjusted turnbuckle combination shall have no more than three threads of the terminal, fork, or eye exposed outside the barrel of the turnbuckle.

	Column Forward	Neutral	Column Aft
Pull Force	Step (c) _____	Step (d) _____	Step (e) _____
Release Force	Step (h) _____	Step (g) _____	Step (f) _____
Breakout Force	Step (i) _____	Step (i) _____	Step (i) _____

Elevator Friction Test Chart
Figure 202

(6) Remove rigging pin and check elevators for proper operation. (Refer to Inspection/Check, this section.)
(7) Install locking clips on turnbuckles.
(8) Install all previously removed access covers.
(9) Restore aircraft to normal.

3. Inspection/Check

NOTE: Angular dimensions for checking elevator deflections are obtained using a contour plate and a clinometer. (Refer to Adjustment/Test, this section, for checking angular deflection.)

A. Check Elevators for Proper Operation
(1) During operational check, observe response and freedom of movement of control columns and elevators.
(2) Pilot and copilot control columns are one piece and will be synchronized.
(3) Push pilot control column forward from neutral.
(4) Observe the movement of the elevators in a downward direction.
(5) With the autopilot engaged, relax force applied to the control column and observe elevator movement back to the neutral position.
(6) Pull pilot control column back from neutral.
(7) Observe the movement of the elevators in an upward direction.
(8) With the autopilot engaged, relax force applied to the control column and observe elevator movement back to the neutral position.

B. Perform Elevator Control Cable Tension Check (See Figure 203.)

NOTE:
- Perform elevator control cable tension check in accordance with the current inspection intervals specified in Chapter 5.
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The cable tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to 27-00-00, Adjustment/Test.)

(1) Position elevators to the neutral position.
(2) Remove floorboards and attach tensiometer to elevator trailing edge up and trailing edge down cables. Elevator cable tension's shall be 75 (±5) pounds at 75°F (34 [±2.3] kg at 24°C) .

NOTE:
- The elevator cable must be lying against the back of the tensiometer when pressure is applied so that no extra pull is applied to the cable.
- Avoid pushing or pulling on tensiometer or cable while a reading is being taken.

(3) Perform Elevator Control Cable Tension Adjustment if readings do not fall within tolerances. (Refer to Adjustment/Check, this section.)
(4) Remove tensiometer from elevator control cable and install floorboards.
(5) Restore aircraft to normal.

ELEVATOR DOWNSPRING - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Caging Pin		Local Manufacture	CagingDownspring.
Low Temperature Grease MIL-G-21164	Aeroshell 17	Shell Houston, TX	Lubricate bolts and bushings.
Low Temperature Grease	Royco 64	Royal Lubricants Dallas, TX	Lubricate bolts and bushings.
Low Temperature Grease MIL-G-21164	Braycote 664	Castrol Commerce, CA	Lubricate bolts and bushings.

B. Removal of Down-spring (See Figure 201.)
   (1) Remove down-spring assembly access covers from vertical stabilizer to gain access to elevator downspring installation.
   (2) Move elevators until caging pin hole is exposed.

NOTE: Caging pin is fabricated from 3/16 inch (4.76 millimeter) round steel stock. Make 90° bend, approximately 3 inches (7 centimeters) from end of 6 inch (15 centimeter) piece of stock. Paint one end of piece red (for visibility).

   (3) Insert caging pin through drawbolt.

WARNING: THE EXISTING SPRING TENSION OF THE CAGED DOWN-SPRING ASSEMBLY IS APPROXIMATELY 300 POUNDS (136 KG). USE EXTREME CARE WHEN REMOVING ATTACHING PARTS AND DOWN-SPRING ASSEMBLY FROM AIRCRAFT.

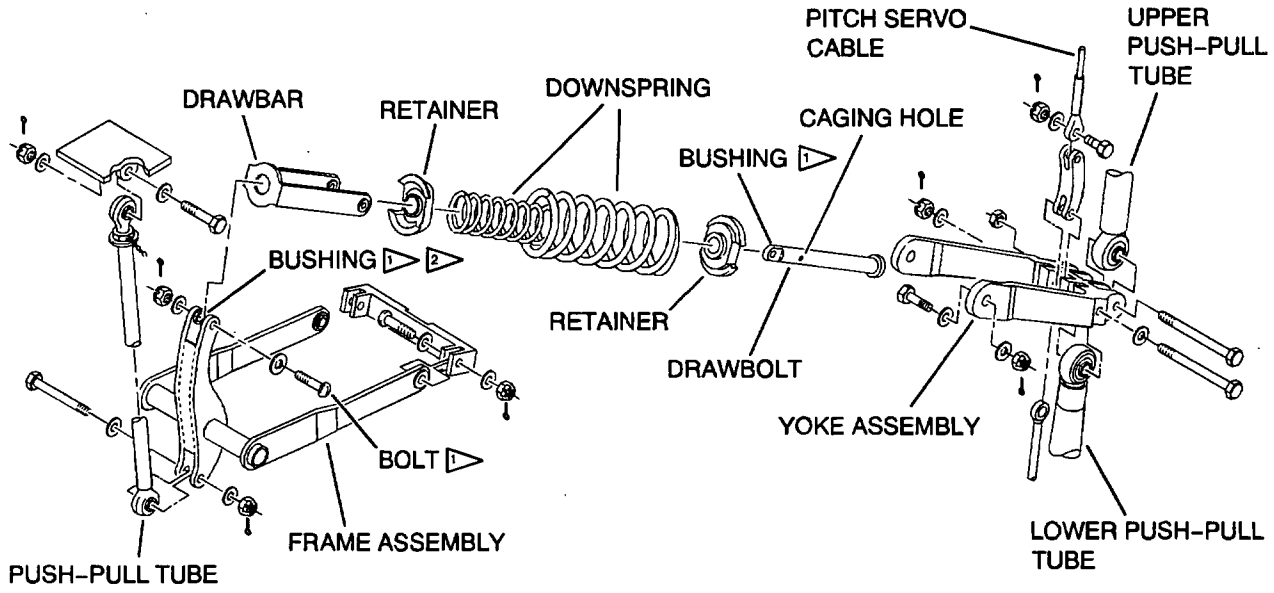
   (4) Remove attaching parts securing forward end of down-spring assembly to frame assembly.
   (5) Loosen tension on pitch servo cables.
   (6) Remove attaching parts securing pitch servo cables and aft end of down-spring assembly to yoke assembly. Remove down-spring assembly from aircraft.

C. Installation of Down-spring (See Figure 201.)

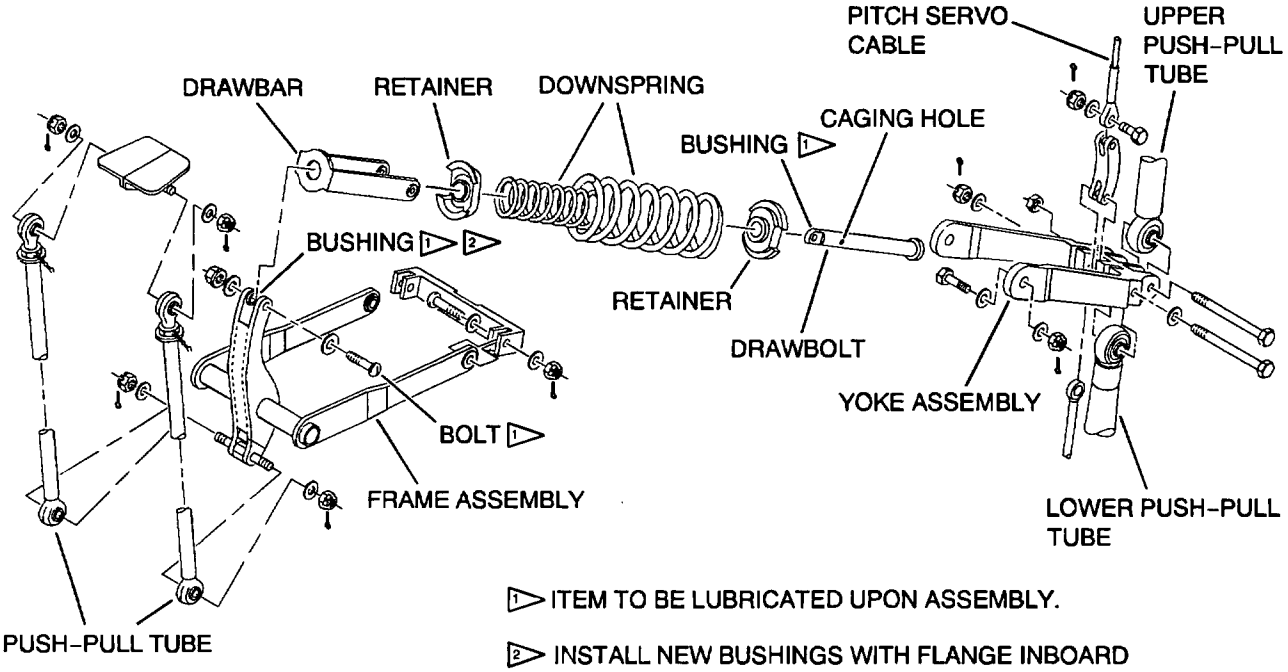
NOTE: When installing the down-spring assembly, use grease in large enough quantities to completely cover the down-spring assembly to frame assembly attach bolt and bushings to protect them from contaminants.

   (1) Install down-spring assembly and pitch servo cables and secure with attaching parts.
   (2) Secure forward end of spring assembly to frame assembly with attaching parts.

WARNING: DO NOT ATTEMPT TO REMOVE THE CAGING PIN UNLESS DOWN-SPRING ASSEMBLY IS COMPRESSED AND BOTH ENDS OF THE DOWN-SPRING ASSEMBLY ARE RESTRAINED.



35-002 THRU 35-041 AND 36-002 THRU 36-016 NOT MODIFIED PER AMK 75-10, "INSTALLATION OF DUAL ELEVATOR UP/DOWN SPRING SECONDARY RETENTION PUSHRODS".



1 ITEM TO BE LUBRICATED UPON ASSEMBLY.

2 INSTALL NEW BUSHINGS WITH FLANGE INBOARD AND SECURE WITH LOCTITE. AFTER CURE REAM IN LINE TO 0.1895 (+0.0010/-0.0000) INCH.

35-042 AND SUBSEQUENT, 36-017 AND SUBSEQUENT, AND PRIOR AIRCRAFT MODIFIED PER AMK 75-10, "INSTALLATION OF DUAL ELEVATOR UP/DOWN SPRING SECONDARY RETENTION PUSHRODS".

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14-42C-1

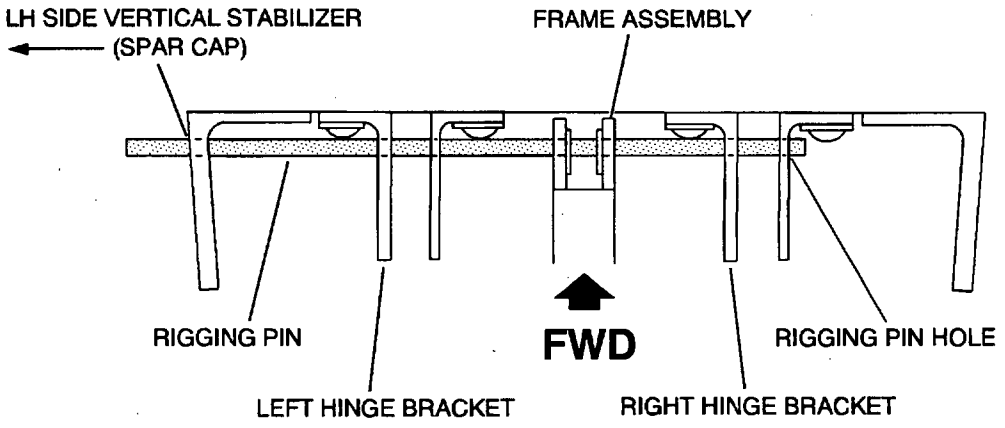
Elevator Downspring Installation
Figure 201

(3) Adjust elevator downspring. (Refer to Adjustment/Test, this section.)
(4) Install access covers on vertical stabilizer.

2. Adjustment/Test
A. Adjust Elevator Downspring Assembly (See Figure 202.)
(1) Move elevators until downspring caging pin hole is exposed.

WARNING: THE EXISTING SPRING TENSION OF THE CAGED DOWNSPRING ASSEMBLY IS APPROXIMATELY 300 POUNDS (136 KG). USE EXTREME CARE WHEN REMOVING ATTACHING PARTS AND DOWNSPRING ASSEMBLY FROM AIRCRAFT.

(2) Insert downspring caging pin and remove bolt and nut attaching drawbolt to frame assembly.
(3) Position horizontal stabilizer to 5° 30' (±0.0° 15') leading edge down from neutral using the horizontal stabilizer rigging plate. (Refer to 27-40-00, Adjustment/Test.)
(4) Remove screw and nut from LH side of vertical stabilizer (spar cap).
(5) Insert rigging pin thru spar cap hole and into LH yoke assembly bracket.
(6) Remove safety wire, loosen jamnut(s), and adjust downspring push-pull tube(s) until rigging pin can be inserted thru the frame assembly and into the right bracket as shown.
(7) Tighten downspring push-pull tube jamnut(s) and safety wire. Remove rigging pin and install screw and nut in spar hole. Reattach drawbolt to frame assembly and remove caging pin.
(8) Perform elevator neutral position and travel adjustment. (Refer to 27-30-03, Adjustment/Test.)
(9) Return aircraft to normal.



(VIEW LOOKING DOWN)

M35-273001-202-01

Elevator Downspring Adjustment
Figure 202

2. Inspection/Check

NOTE: Disassemble and inspect down-spring assembly and attaching parts in accordance with the current inspection intervals specified in Chapter 5.

A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Hole Micrometer		Commercially Available	Measuring hole dimensions.

B. Disassemble and Inspect Down-Spring Assembly

(1) Remove down-spring assembly and frame assembly. (Refer to Removal/Installation, this section.)

WARNING: THE EXISTING SPRING TENSION OF THE CAGED DOWNSPRING ASSEMBLY IS APPROXIMATELY 300 POUNDS (136 KG). USE EXTREME CARE WHEN REMOVING ATTACHING PARTS AND DOWN-SPRING ASSEMBLY FROM AIRCRAFT.

(2) Restrain and compress down-spring assembly and remove caging pin. Carefully release spring tension.
(3) Visually inspect drawbolt for wear or damage. If wear or damage exists, the bolt must be replaced.
(4) Remove bushing from drawbolt. Install new bushing. If drawbolt bushing hole out-of-round exceeds 0.005 inch (0.127 mm), the drawbolt must be replaced.
(5) Visually inspect drawbar and bushings. If drawbar and bushing are worn or damaged they must be replaced.
(6) When assembling the down-spring assembly, use grease in large enouph quantity to completely cover the drawbolt assembly and bushing to protect them from contaminants.
(7) Assemble and compress down-spring assembly and install caging pin.
(8) Install new bushings (2 ea.) in frame assembly with Loctite flange side in. When the Loctite has cured, line ream bushings to 0.1895 (+0.0010/-0.0000) inch [4.81 mm]. Install frame assembly. If frame assembly bushing hole out-of-round exceeds 0.005 inch (0.127 mm), the frame assembly must be replaced.
(9) Visually check all bolts, bushings and assemblies associated with the down-spring assembly.
(10) Install down-spring assembly. (Refer to Removal/Installation, this section.)

CONTROL COLUMN ASSEMBLY - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Rigging Pin		Local Manufacture	Rigging

2. Removal/Installation

A. Remove Control Column Assembly (See figure 201.)

(1) Open tailcone access door and remove electrical power from aircraft.

(2) Remove applicable cockpit seating, carpeting, and floorboards as required to gain access to elevator forward sector assembly, elevator sector push-pull tube, and control column assembly.

(3) Remove both forward and aft pedestal assemblies in accordance with procedures outlined in Chapter 31.

NOTE: Rigging pin is fabricated from 3/16 inch (4.76 millimeter) round steel stock. Make 90° bend, approximately 8 1/2 inches (21 centimeters) from end of 12 inch (30.5 centimeter) piece of stock. Paint short end of piece red (for visibility).

(4) Install a rigging pin in elevator forward sector assembly.

(5) Gain access to aileron sector and install a rigging pin.

NOTE: On *Aircraft 35-108 and Subsequent and 36-032 and Subsequent*, a hole is provided in the keel beam access cover approximately six inches (15.24 centimeters) forward of frame 22. This hole allows the rigging pin to be inserted in the aileron sector without removal of the access panel.

(6) Remove applicable aileron control cables from control wheels in accordance with procedures outlined in appropriate section of this chapter.

(7) Disconnect electrical plug (located under pedestal assemblies) extending to both left-hand and right-hand side of control column.

(8) Identify and tag wires extending to left-hand and right-hand side of control column from plug. Extract wires from plug with pin extraction tool.

(9) Remove attaching parts securing elevator sector push-pull tube to control column.

(10) Remove attaching parts securing bob weight push-pull tube to control column.

(11) Remove attaching parts securing left-hand and right-hand center floorboard supports.

NOTE: The aft screws on the left-hand and right-hand center floorboard supports are shorter than the forward support screws. This is to allow adequate clearance for the control column assembly. Tag the aft screws, to ensure proper reinstallation of the left-hand and right-hand center floorboard supports.

(12) Remove attaching parts securing center bearing block assembly to keel beam.

(13) Remove and discard both left-hand and right-hand spring pins from pivot pin.

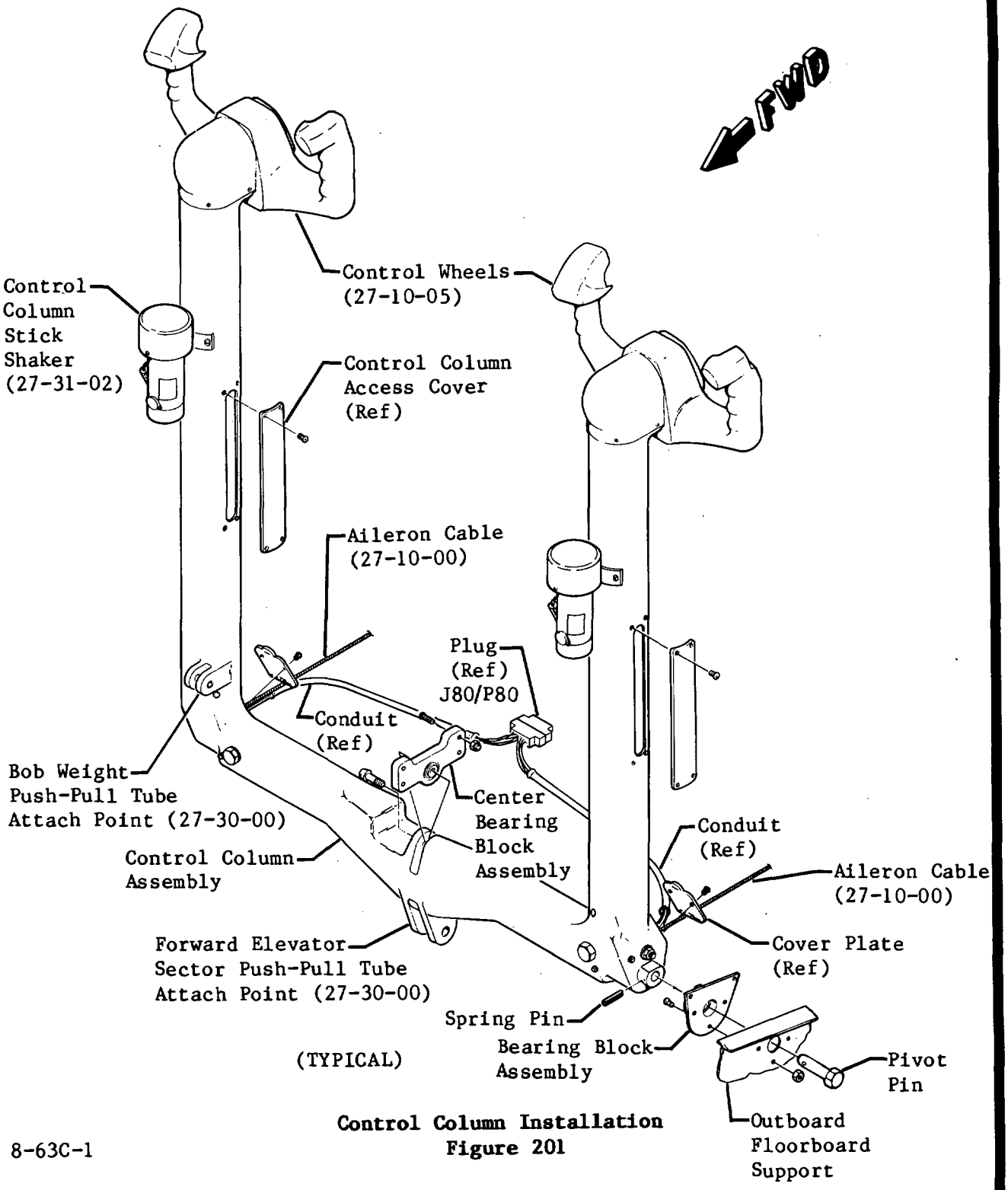
(14) Remove attaching parts securing both left-hand and right-hand bearing block assemblies to outboard floorboard supports.

(15) Remove both left-hand and right-hand bearing block assemblies, bearings, and pivot pins from left-hand and right-hand outboard floorboard supports and control column.
(16) Remove control column assembly from aircraft.

B. Install Control Column Assembly (See figure 201.)
(1) Position control column assembly so as to straddle keel beam.
(2) Position center bearing block assembly and secure to keel beam with attaching parts.
(3) Inspect bearing block assemblies, bearings, and pivot pins for wear, binding, or other possible damage that may inhibit proper operation.
(4) Assemble and position both left-hand and right-hand bearing block assemblies, bearings, and pivot pins to outboard floorboard supports and control column.
(5) Insert new spring pins in pivot pins.

CAUTION: ENSURE SHORTER LH AND RH CENTER FLOORBOARD SUPPORT SCREWS ARE PROPERLY INSTALLED TO ALLOW CLEARANCE OF CONTROL COLUMN ASSEMBLY THROUGH FULL TRAVEL RANGE.

(6) Install both left-hand and right-hand center floorboard supports.
(7) Position elevator sector push-pull tube to control column and secure with attaching parts.
(8) Position bob weight push-pull tube to control column and secure with attaching parts.
(9) Identify wires from left-hand and right-hand side of control column, route wiring, insert wire/pins into electrical plug, and reconnect plug.
(10) Install applicable aileron control cables on control wheels in accordance with procedures outlined in appropriate section of this chapter.
(11) Rig aileron control control system. (Refer to Adjustment/Test, 27-10-00.)
(12) Remove rigging pin from aileron sector. Install access panel, if removed.
(13) Test aileron control system for proper operation. (Refer to Adjustment/Test, 27-10-00.)
(14) Level the aircraft. (Refer to Chapter 8.)
(15) Adjust forward rod end on elevator push-pull tube assembly to obtain correct control column neutral position. *On Aircraft 35-002 thru 35-020 and 36-002 thru 36-013*, control column neutral position is 3° (±1°) aft of vertical. *On Aircraft 35-021 and Subsequent, 36-014 and Subsequent*, control column neutral position is 5° (±0.5°) aft of vertical.
(16) Remove rigging pin from elevator forward sector assembly.
(17) Operationally check elevator control system. (Refer to Inspection/Check, 27-30-00.) Check bob weight clears aircraft structure thru full travel range.
(18) Install floorboards and carpeting.
(19) Install both forward and aft pedestal assemblies (Refer to Chapter 31) and install cockpit seating (Refer to Chapter 25).
(20) Restore aircraft to normal.
(21) Restore electrical power to aircraft and close tailcone access door.



Control Column Installation
Figure 201

8-63C-1

EFFECTIVITY: ALL
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ELEVATOR - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: Removal/Installation procedures are typical for both right and left elevators and elevator tip covers.

For maintenance of the elevator tip covers, refer to Chapter 55.

A. Tools and Equipment

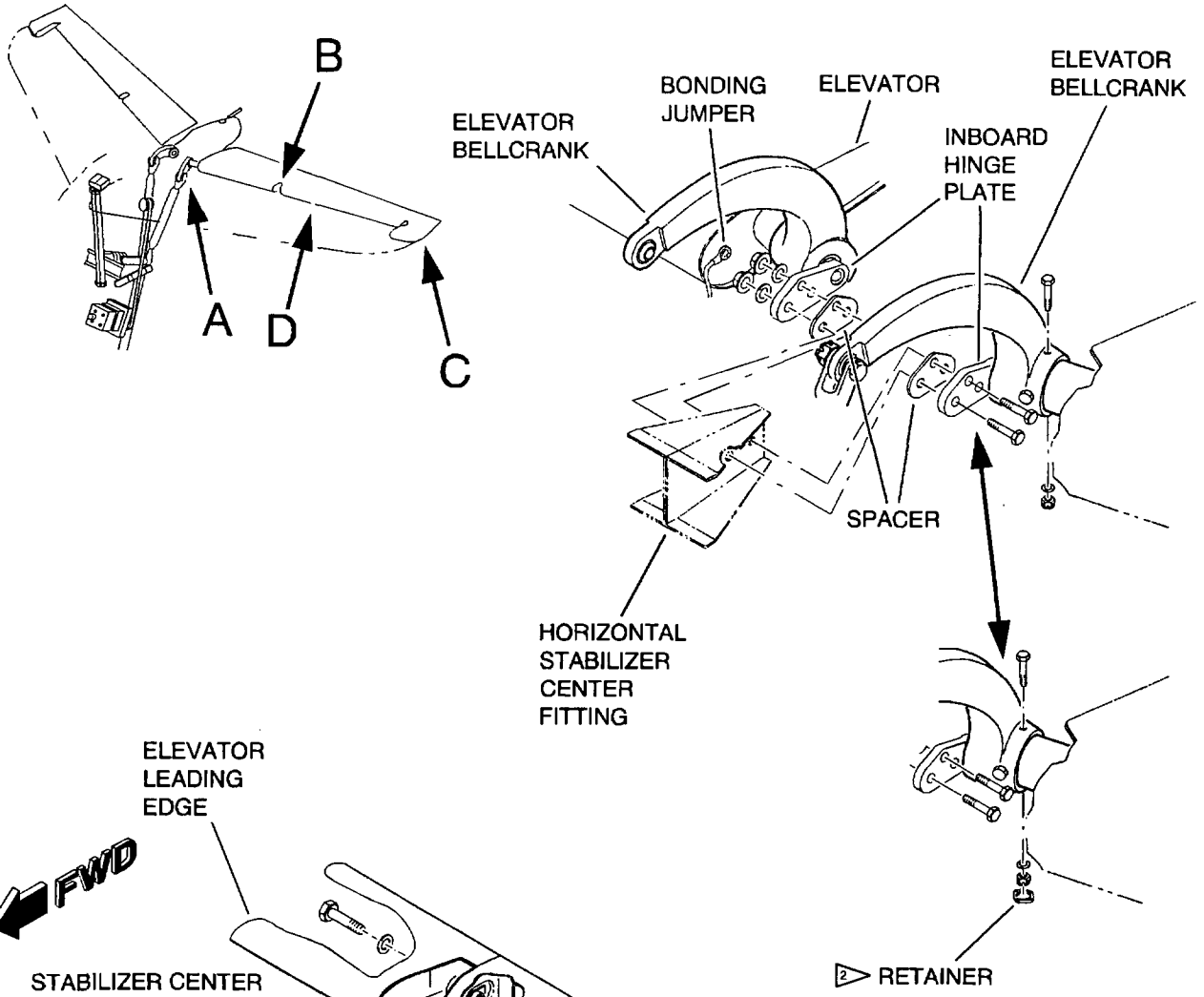
NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Rigging Pin	3170037-12	Learjet Inc. Wichita, KS	Measure gap at horizontal stabilizer and elevator horn.
Rigging Pin	3170037-1	Learjet Inc. Wichita, KS	Hold the elevators in neutral.
Safety Wire	MS20995C32	Commercially Available	Attach the nut retainers.
Heat Shrink	RNF-100-3/16-CL	Raychem Menlo Park, CA	Protect elevator bellcrank.
Torque Wrench		Commercially Available	Torque nuts.

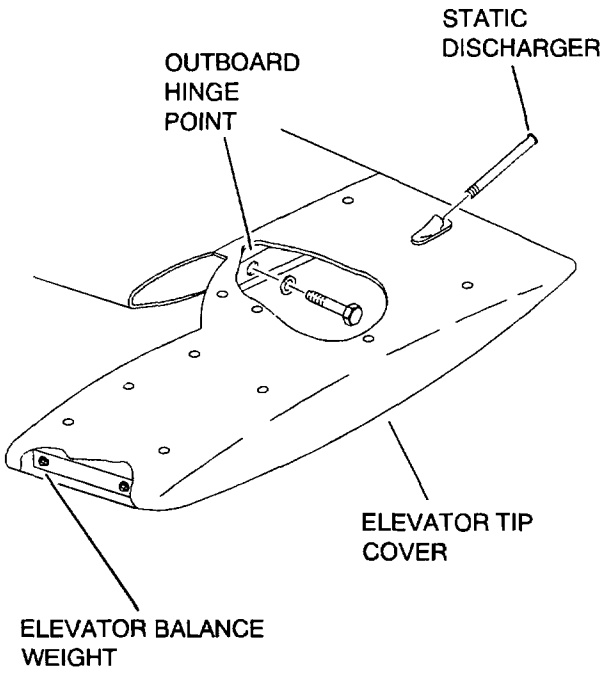
B. Removal of Elevator (See Figure 201.)
(1) Remove access cover and insert rigging pin into aft elevator sector.
(2) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.
(3) Remove aft vertical stabilizer tip access cover.
(4) Remove center hinge access door and elevator tip cover.
(5) Remove bolt connecting elevator bellcrank and upper push-pull tubes.
(6) Remove inboard, center, and outboard hinge bolts from elevator and disconnect bonding jumper on inboard end of elevator.
(7) *On Aircraft 35-292 and Subsequent and 36-045 and Subsequent*, disconnect bonding jumper on center elevator hinge.
(8) Remove elevator from aircraft.

C. Installation of Elevator (See Figure 201.)

NOTE: If elevator is being replaced, install per SSK 995.

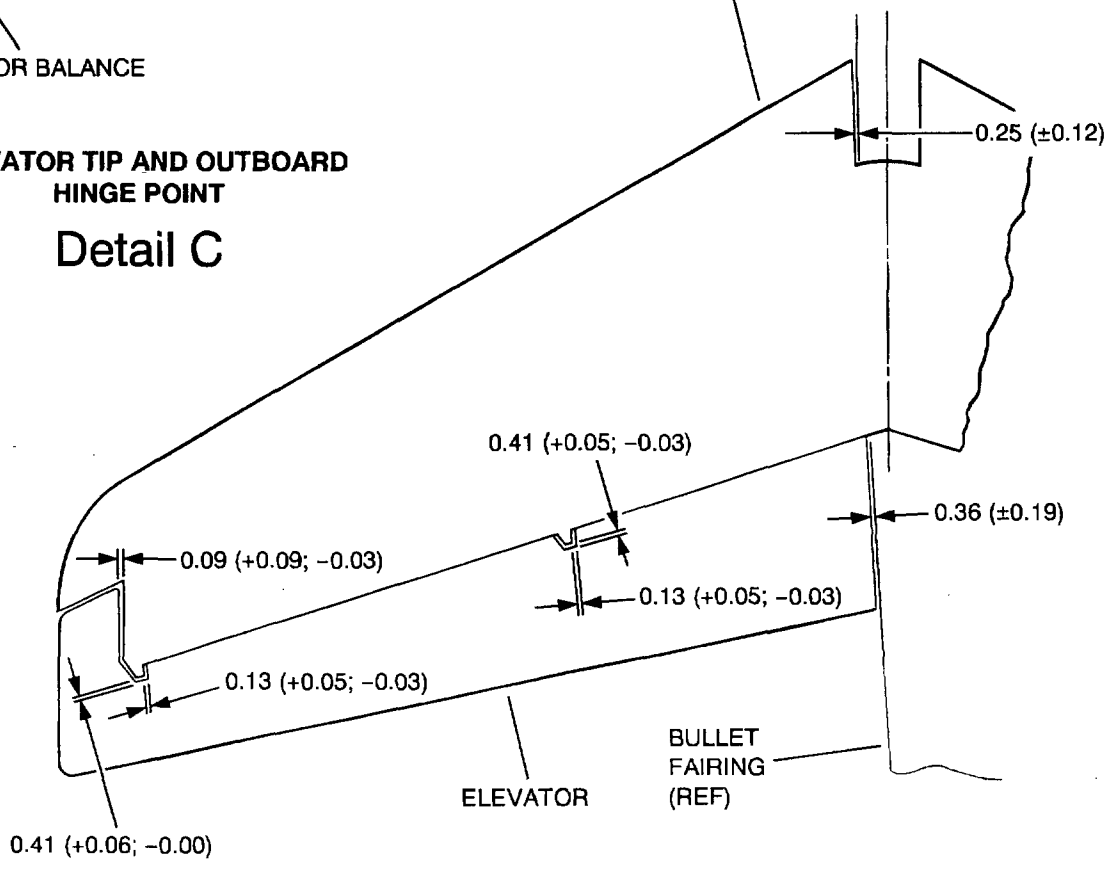


Elevator Installation
Figure 201 (Sheet 1 of 2)



ELEVATOR TIP AND OUTBOARD HINGE POINT

Detail C



ELEVATOR CLEARANCES (VIEW LOOKING DOWN)

Detail D

M35-273003-201-02

Elevator Installation
Figure 201 (Sheet 2 of 2)

(1) Ensure that elevator is properly balanced. (Refer to 27-00-00, Adjustment/Test.)
(2) If elevator bellcrank assembly is not installed, install as follows:
   (a) Install elevator bellcrank assembly and install attaching parts. Torque nuts to 95 to 115 inch pounds [10.7 to 12.9 Nm].
   (b) *On Aircraft 35-002 thru 35-676 and 36-002 thru 36-063 modified per SB 35/36-27-35*, install nut retainers, start safety wire at one end then install heat shrink over safety wire to protect bellcrank. Finish safety wiring nut retainer to bolt.

CAUTION: THIS BOLT MUST BE INSTALLED BEFORE THE LH ELEVATOR IS SET AT ITS CORRECT LOCATION ON THE HORIZONTAL STABILIZER. THE ELEVATOR BELLCRANK ASSEMBLY WILL INTERFER WITH THE INSTALLATION OF THIS BOLT AFTER THE ELEVATOR IS IN POSITION ON THE HORIZONTAL STABILIZER.

(3) With the LH elevator on the bench, install an inboard hinge bolt in the lower aft position in the inboard hinge plate.
(4) Install the shim(s) on the inboard hinge bolt.
(5) Set the LH elevator at its correct location on the horizontal stabilizer with the inboard hinge bolt in the horizontal stabilizer hinge.
(6) With the shims(s) correctly aligned, install the remaining inboard hinge bolts. Make sure that the bolts are installed the asme as they were when they were removed.
(7) Install the remainijng elevator hinge bolts (working from inboard to outboard) that attach the elevator to the horizontal stabilizer at the center and outboard hinge points.
(8) Connect bonding jumper to horizontal stabilizer at inboard stabilizer hinge point.
(9) *On Aircraft 35-292 and Subsequent and 36-045 and Subsequent*, connect bonding jumper to center elevator hinge.
(10) Install elevator hinge bolts securing elevator to horizontal stabilizer at the inboard, center, and outboard hinge points.

NOTE: On some aircraft, the inboard hinge bearings may be secured with Hi-Lok fasteners which are not to be reused.

WARNING: MAKE SURE THAT THE BEARING ASSEMBLIES IN THE ELEVATOR BELLCRANKS ARE PROPERLY INSTALLED AND SECURE. A LOOSE BEARING ASSEMBLY CAN CAUSE THE ELEVATOR TO JAM.

(11) Check and make sure that the bearing assemblies, in the elevator bellcranks, are secure.

WARNING: ENSURE GAP BETWEEN HORIZONTAL STABILIZER AND ELEVATOR HORN IS 0.25 (±0.03) INCH [6.35 (±0.76) MM]. ICE MAY ACCUMULATE ON ELEVATOR HORN LEADING EDGE WITH LESS GAP CLEARANCE WHEN FLYING IN ICING CONDITIONS.

(12) Using a rigging pin, check gap between horizontal stabilizer and elevator horn with elevator in neutral position. Gap shall be 0.25 (±0.03) inch [6.35 (±0.76) mm].

(13) Connect elevator bellcrank to upper push-pull tubes in vertical stabilizer.
(14) Install leading edge access cover and elevator tip cover.
(15) Check elevator for proper operation and travel. (Refer to 27-30-00.)
(16) Install aft vertical stabilizer tip access cover.
(17) Install horizontal stabilizer fairings on both sides of vertical stabilizer.

2. Adjustment/Test

A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Clinometer	2471004	Learjet Inc. Wichita, KS	Measure deflection angles of elevator.
Contour Plate	2471002	Learjet Inc. Wichita, KS	Locate control surface neutral.
Rigging Pin		Local Manufacture	Elevator rigging.

B. Elevator Neutral Position and Travel Adjustment (See Figure 202.)
(1) Remove access cover allowing access to elevator aft sector assembly and install rigging pin in elevator aft sector assembly.

NOTE: Rigging pin is fabricated from 3/16 inch [4.76 mm] round steel stock. Make 90 degrees bend, approximately 8 1/2 inches [21 cm] from end of 12 inch [30.5 cm] piece of stock. Paint short end of piece red (for visibility).

(2) Remove access covers located on vertical stabilizer, allowing access to elevator pushrod assemblies.
(3) Install contour plate on horizontal stabilizer at LBL 19.40.

NOTE: In the event that the elevator contour plate makes contact with the elevator fixed trailing tab, it is permissible to modify the plate, to prevent any contact between the plate and tab. See Figure 202 for modification information.

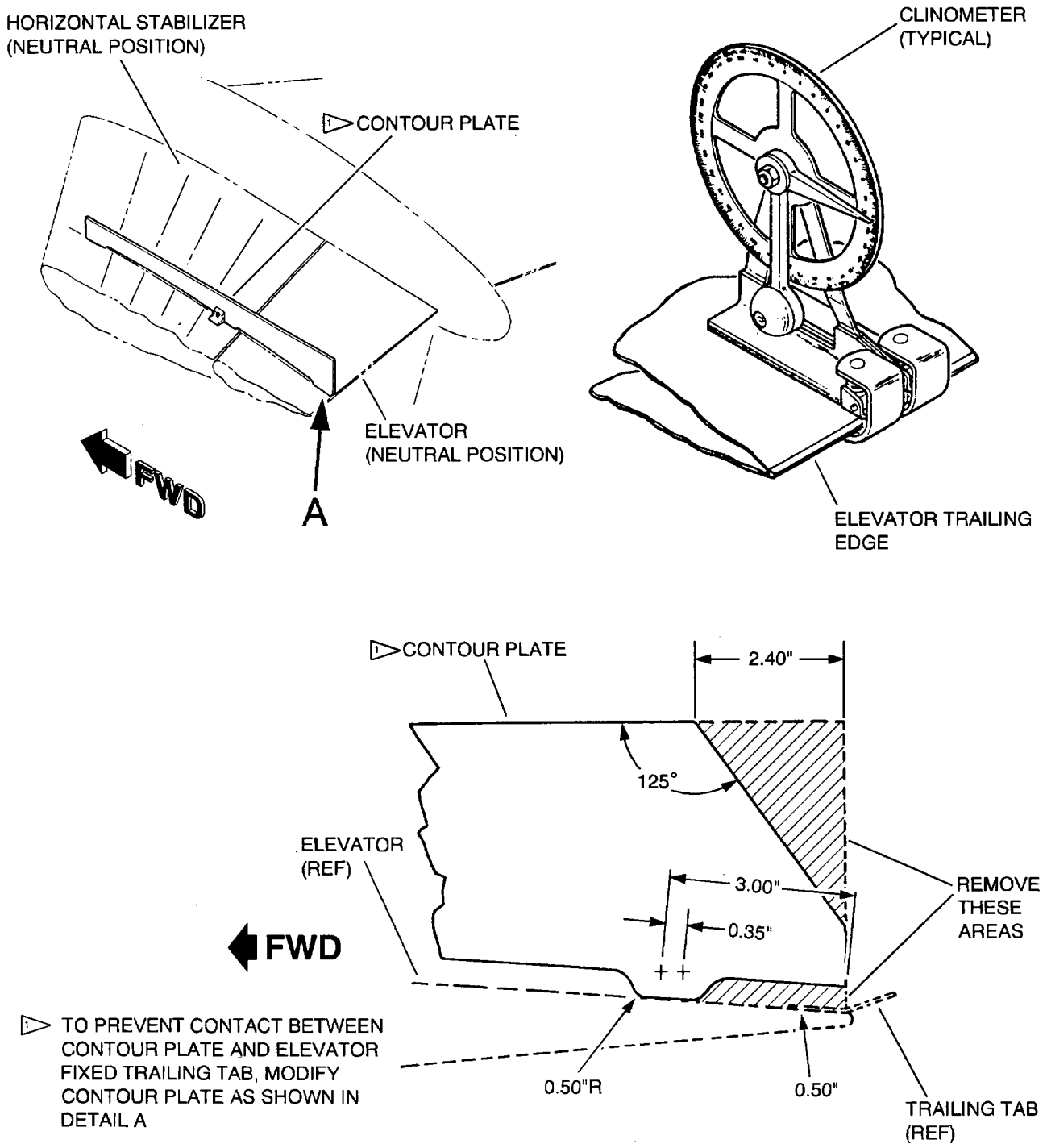
(4) Adjust LH elevator upper pushrod assembly sufficient to fair elevator with horizontal stabilizer.
(5) Repeat steps 2.B.(3) and 2.B.(4) for RH elevator assembly.
(6) Install clinometer on RH elevator assembly and adjust clinometer to 0 degrees.
(7) Remove contour plate from horizontal stabilizer.
(8) Remove rigging pin from elevator aft sector assembly.
(9) Set the primary up stop bolts (located at aft sector in tailcone) to obtain elevator trailing edge up travel of 16 degrees (+0', -30') from the rig position. *On Aircraft 35-040 and Subsequent, 36-017 and Subsequent, and prior Aircraft modified by SB 35/36-27-3*, safety-wire the stop bolt.

(10) Adjust secondary stop bolts on the forward elevator sector to contact their stops when 100 (±25) pounds (45.36 [±11.34] kg) of force is applied to the control column wheel (fore and aft) after primary stop bolts are contacted.
(11) Move control column forward and aft until secondary stops are contacted. Note the position of the bob weight with respect to aircraft structure at each extreme position. Adjust bob weight push-pull tube as required to prevent bob weight from contacting structure.
(12) Adjust bobweight stop bolts so that the secondary stop bolts on the forward sector and the bobweight stop bolts contact their stop simultaneously in either the up or down position.
(13) Remove clinometer from elevator assembly.
(14) Check elevators for proper operation. (Refer to 27-30-00, Inspection/Check.)
(15) Install all previously removed access covers.
(16) Restore aircraft to normal.

3. Repairs

A. Replacement of Elevator Fixed Trailing Tab *(Aircraft with elevators installed per SSK 995.)*
(1) Mark location of elevator fixed trailing tab.
(2) Remove attaching parts securing trailing tab to elevator assembly.
(3) Position new trailing tab at its appropriate location on elevator assembly and mark rivet pattern on tab.
(4) Drill five (5) forward holes #27 (0.144 inch dia. [3.6 mm]) and 100 degrees countersink (100 degrees x 0.225 inch dia. [5.7 mm]).
(5) Drill 12 aft holes #30 (0.128 inch dia. [3.2 mm]).
(6) Position tab on elevator and secure with attaching parts.
(7) Verify angular position of trailing tab.
(a) *On Aircraft 35-002 thru 35-066 and 36-002 thru 36-017*, angular position shall be 12 degree (±2 degrees) up.
(b) *On Aircraft 35-067 and Subsequent, 36-018 and Subsequent, and prior Aircraft modified per AAK 76-4*, angular position shall be 9 degrees (±2 degrees) up.

B. Replacement of the Elevator Bellcrank Bearing
(1) Remove the elevator from the aircraft. (Refer to 27-30-03.)
(2) Remove the attaching parts that attach the bellcrank assembly to the elevator assembly.
(3) Replace the bearing in the bellcrank assembly. (Refer to 20-25-01.)
(4) Install the attaching parts that attach the bellcrank to the elevator assembly. (Refer to 27-30-03.)
(5) Install the elevator on the aircraft. (Refer to 27-30-03.)



M35-273003-202-01

Elevator Neutral Position and Travel Adjustment
Figure 202

ELEVATOR SECTORS (FORWARD/AFT) - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Rigging Pin		Local Manufacture	Rigging Controls
Safetywire	MS20995-F32	Commercially Available	Safetying Parts

2. Removal/Installation

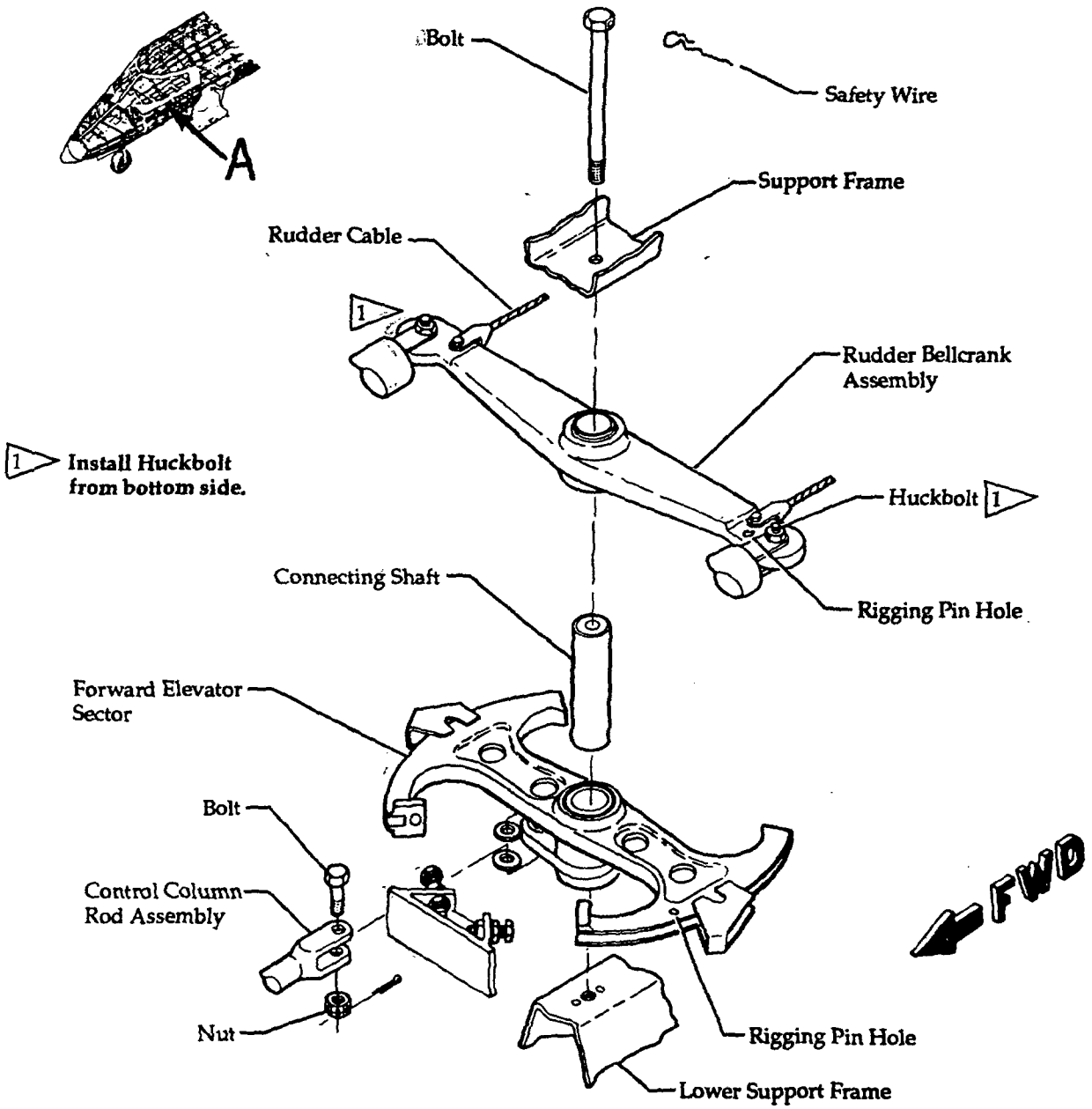
NOTE:
- When forward elevator sector is removed or replaced, the rigging and inspection of the elevator control system (Refer to 27-30-00 Adjustment/Test) and rudder control system (Refer to 27-20-00 Adjustment/Test), must be performed.
- When aft elevator sector is removed or replaced, the rigging and inspection of the elevator control system (Refer to 27-30-00 Adjustment/Test) and adjustment/test of the autopilot elevator position sensor (Refer to 22-10-06 Adjustment/Test), must be performed.

A. Removal Forward Elevator Sector (See figure 201.)
   (1) Remove cabin center aisle floorboards and access panels, as required to gain access to the rudder bellcrank assembly and forward elevator sector.
   (2) Ensure rudder bellcrank assembly and forward elevator sector are in neutral position and insert rigging pin through both.

       NOTE: Rigging pin is fabricated from 0.250 inch round steel stock.

   (3) Remove attaching parts and disconnect rudder connecting rod assemblies *(Aircraft 35-002 thru 35-505 except 35-406, and 36-002 thru 36-053 not modified per AAK 83-2, "Installation of FC-530 Autopilot.")* or force sensors *(Aircraft 35-408, 35-506 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot.".*
   (4) Loosen and disconnect rudder and elevator cables from rudder bellcrank assembly and forward elevator sector.
   (5) Remove rigging pins.
   (6) Cut safety wire, remove attaching bolt, rudder bellcrank assembly and forward elevator sector from lower support frame.
   (7) Disassemble rudder bellcrank assembly, forward elevator sector, and connecting shaft.

B. Install Forward Elevator Sector (See figure 201.)
   (1) Install connecting shaft in elevator sector.
   (2) Assemble rudder bellcrank assembly on connecting shaft and elevator sector.
   (3) Position rudder bellcrank assembly and elevator sector on lower support frame and secure with attaching bolt.
   (4) Safety sector attachment bolt with MS20995-F32 safetywire using double twist method.
   (5) Insert rigging pin through both elevator sector and rudder bellcrank assembly.
   (6) Wrap and connect elevator and rudder cables.



(TYPICAL)

Detail A

8-131B

Forward Elevator Sector Installation
Figure 201

(7) Connect rudder connecting rod assemblies (*Aircraft 35-002 thru 35-505 except 35-406, and 36-002 thru 36-053 not modified per AAK 83-2, "Installation of FC-530 Autopilot."*) or force sensors (*Aircraft 35-408, 35-506 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot."*.

(8) Remove rigging pins.

(9) Perform rigging elevator control system. (Refer 27-30-00, Adjustment/Test.)

(10) Perform rigging rudder control system. (Refer 27-20-00, Adjustment/Test.)

(11) Install all previous equipment removed to gain access to the forward elevator sector.

(12) Return aircraft to normal.

C. Removal Aft Elevator Sector (See figure 202.)

(1) Remove fuselage aft access covers to gain access to the aft elevator sector.

(2) Set aft elevator sector in neutral position and insert rigging pin.

NOTE: Rigging pin is fabricated from 3/16 inch round steel stock.

(3) Remove bolt connecting elevator bellcrank and push-pull tube.

(4) Remove attaching parts from elevator position sensor arm and remove from aft elevator sector.

(5) Loosen and disconnect elevator cable.

(6) Remove rigging pin.

(7) Remove attaching parts and aft elevator sector from aircraft.

D. Install Aft Elevator Sector (See figure 202.)

(1) Position aft elevator sector on mounting bracket and install with attaching parts.

(2) Install rigging pin.

(3) Wrap and connect elevator cable.

(4) Position the autopilot elevator position sensor on aft elevator sector and secure with attaching parts.

(5) Position push-pull tube on aft elevator sector and secure with attaching parts.

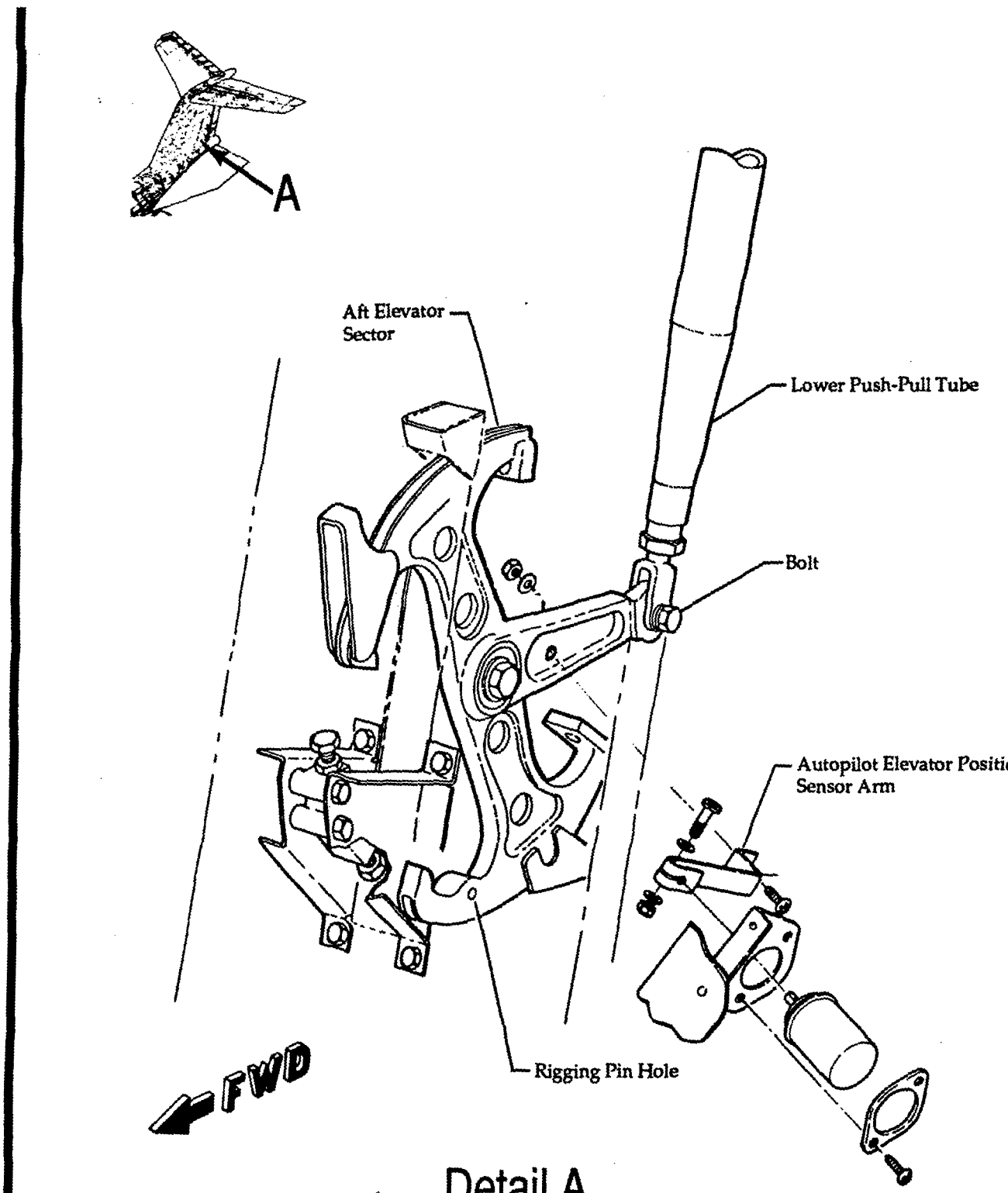
(6) Remove rigging pin.

(7) Perform rigging elevator control system. (Refer 27-30-00, Adjustment/Test.)

(8) Perform adjustment/test of the autopilot elevator position sensor. (Refer to Chapter 22.)

(9) Install fuselage aft access covers.

(10) Return aircraft to normal.



Detail A

Aft Elevator Sector Installation
Figure 202

STALL WARNING - DESCRIPTION AND OPERATION

1. DESCRIPTION

A. The dual stall warning and pusher control system provides the crew with an indication of an impending stall and also commands an aircraft nose down attitude as corrective action.

B. On Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System and not modified per AAK 83-2, "Installation of FC-530 Autopilot," the stall warning system consists of two angle-of-attack transducers, a stall warning computer-amplifier, a stall warning bias box, two control column shakers, a normal accelerometer, an angle-of-attack indicator, two ON-OFF control switches, a stall warning accelerometer cutout box, and two power-on warning lights. The system also utilizes the elevator pitch servo.

C. On Aircraft 35-067 and Subsequent and 36-018 and Subsequent and prior aircraft equipped with Reduced Approach Speed System, the stall warning system consists of two angle-of-attack transducers, a stall warning computer-amplifier, two control column shakers, a normal accelerometer, two angle-of-attack indicators, two ON-OFF control switches, two altitude switches, and two power-on stall warning lights. The stall system utilizes the Test Switches on the test switch panel for testing and also utilizes the elevator pitch servo.

D. **Components Description** (See figure 1.)

(1) An angle-of-attack transducer is installed on each side of the fuselage forward of the forward pressure bulkhead at stringer 12. Access to the transducers is through the nose compartment access doors. An angle-of-attack vane is attached to each transducer.

(2) The stall warning computer-amplifier is installed on the top forward side of frame 2.

(3) On Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System and not modified per AAK 83-2, "Installation of FC-530 Autopilot," a stall warning bias box is installed forward of frame 3 adjacent to the stall warning accelerometer cutout box.

(4) A control column shaker is installed on each control column.

(5) On Aircraft 35-002 thru 35-505 and 36-002 thru 36-053, the normal accelerometer is installed on the aft LH side of frame 2.

(6) An angle-of-attack indicator is installed in the pilot's instrument panel. On Aircraft 35-067 and Subsequent, 36-018 and Subsequent, an angle-of-attack indicator is installed in both crew members' instrument panel.

(7) The ON-OFF control switches are the STALL WARNING Switches located on the test switch panel.

(8) On Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System, a stall warning accelerometer cutout box is installed forward of frame 3 between the stall warning bias box and the air data sensor.

(9) On Aircraft 35-067 thru 35-505, 36-018 thru 36-053 not modified per AAK 83-2, "Installation of FC-530 Autopilot," the altitude switches are installed in the pitot-static plumbing between frames 4 and 5. On Aircraft 35-506 and Subsequent and 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot," the altitude switches are part of the airspeed/mach indicators on the pilot's and copilot's instrument panels. (Refer to Chapter 34.)

(10) The two power-on warning lights, L STALL and R STALL, are located on the glareshield.

(11) On Aircraft 35-505 and Subsequent and 36-054 and Subsequent, the 1/2 g accelerometer monitor box is located beneath the copilot's seat.

2. OPERATION (See figures 2 thru 5)

A. On Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System and not modified per AAK 83-2, "Installation of FC-530 Autopilot," the angle-of-attack transducers are supplied with a regulated voltage which is modified by the stall warning bias box. The wiper of the angle-of-attack transducer supplies a voltage to the computer-amplifier which is proportional to the aircraft angle of attack. When the aircraft angle of attack increases such that speed is 7% above aircraft stall speed, the stall warning computer-amplifier energizes the control column shaker motors. The shaker motors then produce a low-frequency, high-amplitude buffet signal to the crew members. When both angle-of-attack transducer vanes simultaneously increase to 5% above stall condition, the computer-amplifier energizes the pitch servo to command an aircraft nose down direction. The amount of force applied is 80 pounds at the control wheel. The stall warning accelerometer cutout completes a holding circuit and prevents any decrease in the nose down force by the normal accelerometer. As soon as the angle-of-attack transducer vanes decrease below the stall point, the nose down command is removed. In the event one angle-of-attack transducer vane increases 5% above a stall condition, due to wind gust or bird strike, the shaker will energize and the computer amplifier will command a nose down attitude. The amount of force applied is 50 pounds. This force diminishes linearly through the normal accelerometer to maintain 1/2 "g" downward push at the control wheel. The angle-of-attack indicator provides continuous readout of the aircraft angle of attack. Normal autopilot information is first fed into the stall warning computer-amplifier and then to the elevator pitch servo. When the angle-of-attack transducer vanes increase to the pusher actuation point, the autopilot information is disconnected from the pitch servo. The stall warning system is powered by the battery bus.

B. On Aircraft 35-067 thru 35-505, 36-018 thru 36-053, and earlier aircraft equipped with Reduced Approach Speed System, but not modified per AAK 83-2, "Installation of FC-530 Autopilot," the angle-of-attack transducers are supplied with a constant voltage. As the angle-of-attack changes, the angle-of-attack vanes move, causing the transducers' signal to the stall warning computer to change. The stall warning computer compares the angle-of-attack transducer signal and flap position with altitude information to determine if the aircraft is approaching an unsafe flight attitude. Flap

position information is fed to the computer from a three-layered position switch at each flap sector. The flap position switches provide the computer with three different flap positions: 0° to 3°, 0° to 13°, 0° to 25°. Above 22,500 (±750) feet, the altitude switches close. When the altitude switches are closed, the control column shaker and computer pusher function actuation points are approximately 15 knots above stall speed. Below 22,500 (±750) feet, the shakers produce a low- frequency, high-amplitude buffet transmitted through the control column when the airspeed falls to within 7% of stall speed. As the shakers actuate, the angle-of-attack indicator needles enter the yellow arc and the stall warning lights will illuminate and flash. If the pending stall condition continues until the angle-of-attack indicator reaches its red line, the computer will command a nose down attitude (pusher function) to the pitch servo within one knot of stall speed. The amount of force applied at the control wheel is 80 pounds (pusher function) and is continued in a pumping motion until the pending stall condition is resolved. If the pitch attitude change is abrupt, the accelerometer will limit the pusher function so that aircraft g forces will not go below +0.55 g. (The accelerometer also limits the computer pitch command in the event of a computer malfunction.) Should a malfunction occur in the stall warning system, the stall warning lights will illuminate continuously, not flash. To test the system, set Test Switch to the desired position (R STALL or L STALL), depressing the pushbutton simulates an increasing angle of attack which causes the appropriate indicator needle to begin a slow sweep across the green band. As the needle crosses into the yellow band, the control column shaker will begin to operate and the warning light and master warning light will begin to operate and the warning light and master warning light will begin to flash. As the needle reaches the red line, the pusher function will actuate briefly, then stop. The warning light will illuminate steady just prior to or at pusher actuation. The master warning light will continue to flash. After pusher stops, the needle will sweep back and remain in the yellow or green areas.

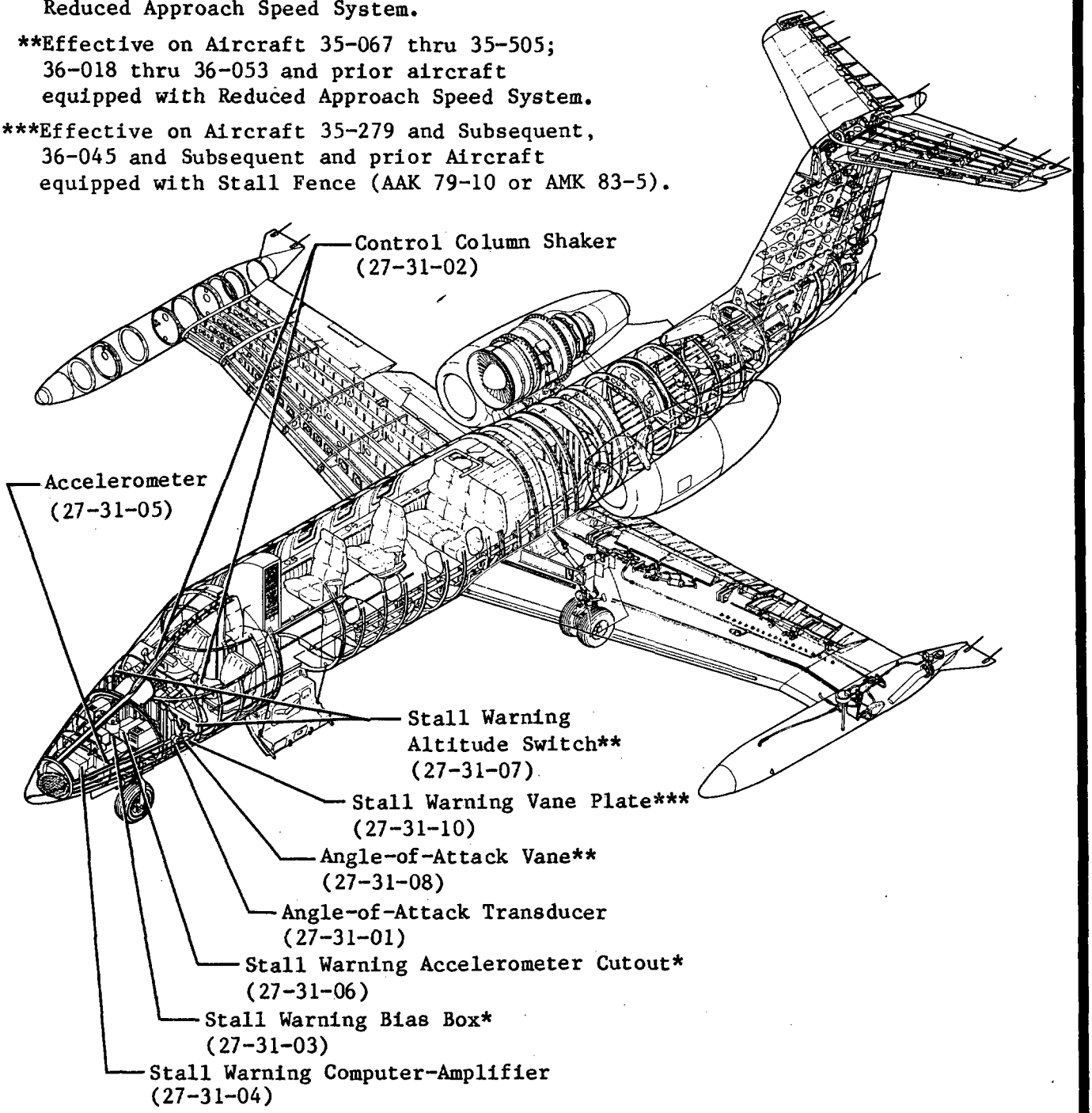
C. On Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot," as angle of attack changes, the stall vanes move, causing the angle-of-attack transducers to signal the stall warning computer. The transducer signal is summed in the stall warning computer along with flap position and altitude information. Flap position information is sent to the computer from position switches at each flap sector. The switches provide the computer with four different flap positions; 0° to 3°, 3° to 13°, 13° to 25°, and 25° to 40°. Below 22,500 (±500) feet altitude, pusher actuation (pusher speed) occurs within ±3 knots of stall speed. Refer to Airplane Flight Manual Stall Speeds Chart. Shaker actuation is at least 7 knots or 7% above pusher actuation. At 22,500 (±500) feet altitude and above, the altitude switches are closed, causing both the pusher and shaker actuation to be increased approximately 15 knots. The shakers produce a high-frequency, low-amplitude buffet transmitted to control columns. As the shakers actuate, the angle-of-attack indicator needle enters the yellow arc, the nudger activates, and the stall warning lights illuminate and flash. At the same time, the nudger detector is armed to sound an aural warning if

the pitch servo does not actuate. If the pending stall condition continues until the angle-of-attack indicator reaches its red line, the computer commands a nose-down attitude (pusher) to the pitch servo. If the pitch servo does not actuate, the nudger detector allows an aural warning to sound. The aural warning continues until the pitch servo is actuated. The amount of force applied at the control wheel is approximately 50 pounds and is continued until the stall condition is resolved. The pusher is limited by separate RH and LH accelerometers so that aircraft g forces will not go below +0.55 g. (The accelerometer also limits the aircraft in the event of a computer malfunction.)

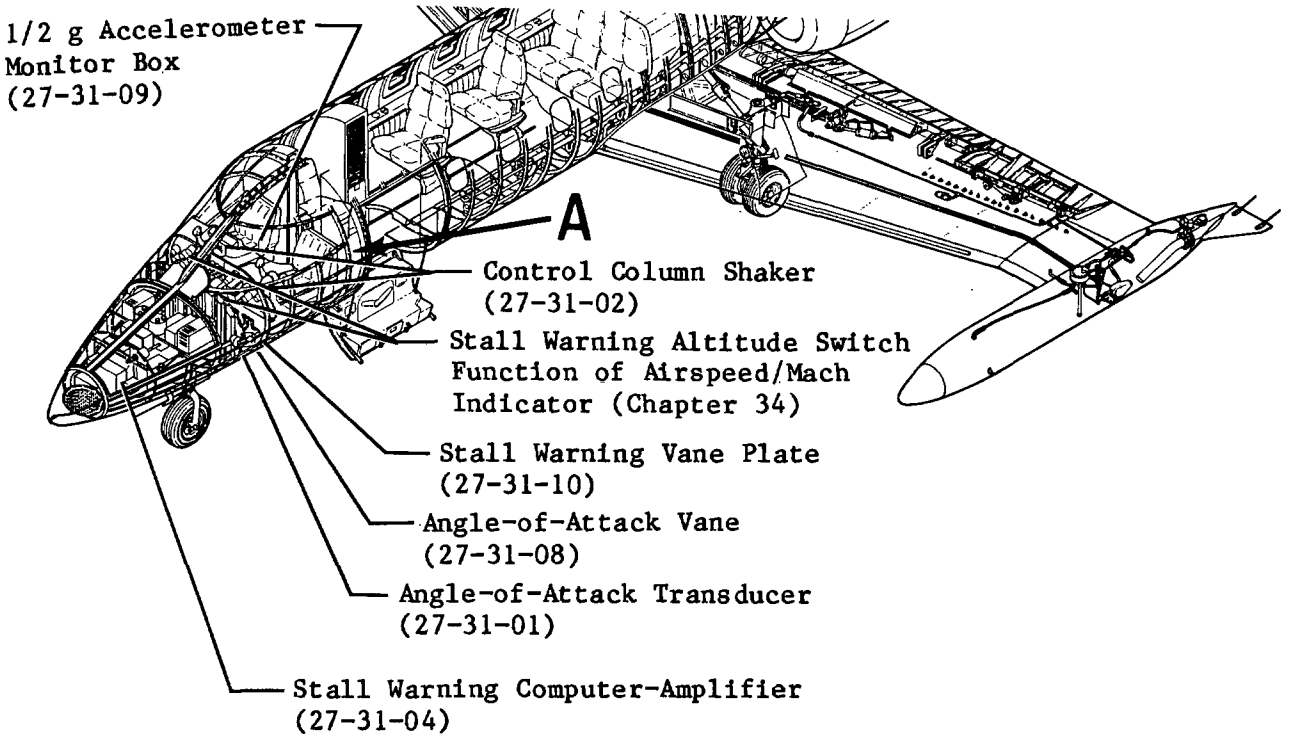
*Effective on Aircraft 35-002 thru 35-066, 36-002 thru 36-017 not equipped with Reduced Approach Speed System.

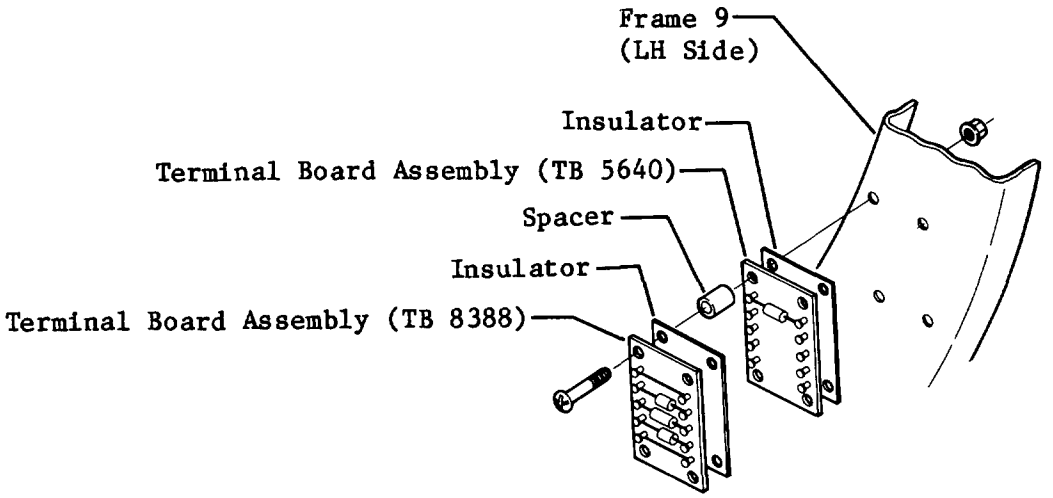
**Effective on Aircraft 35-067 thru 35-505; 36-018 thru 36-053 and prior aircraft equipped with Reduced Approach Speed System.

***Effective on Aircraft 35-279 and Subsequent, 36-045 and Subsequent and prior Aircraft equipped with Stall Fence (AAK 79-10 or AMK 83-5).



Stall Warning System Component Locator
Figure 1 (Sheet 1 of 3)

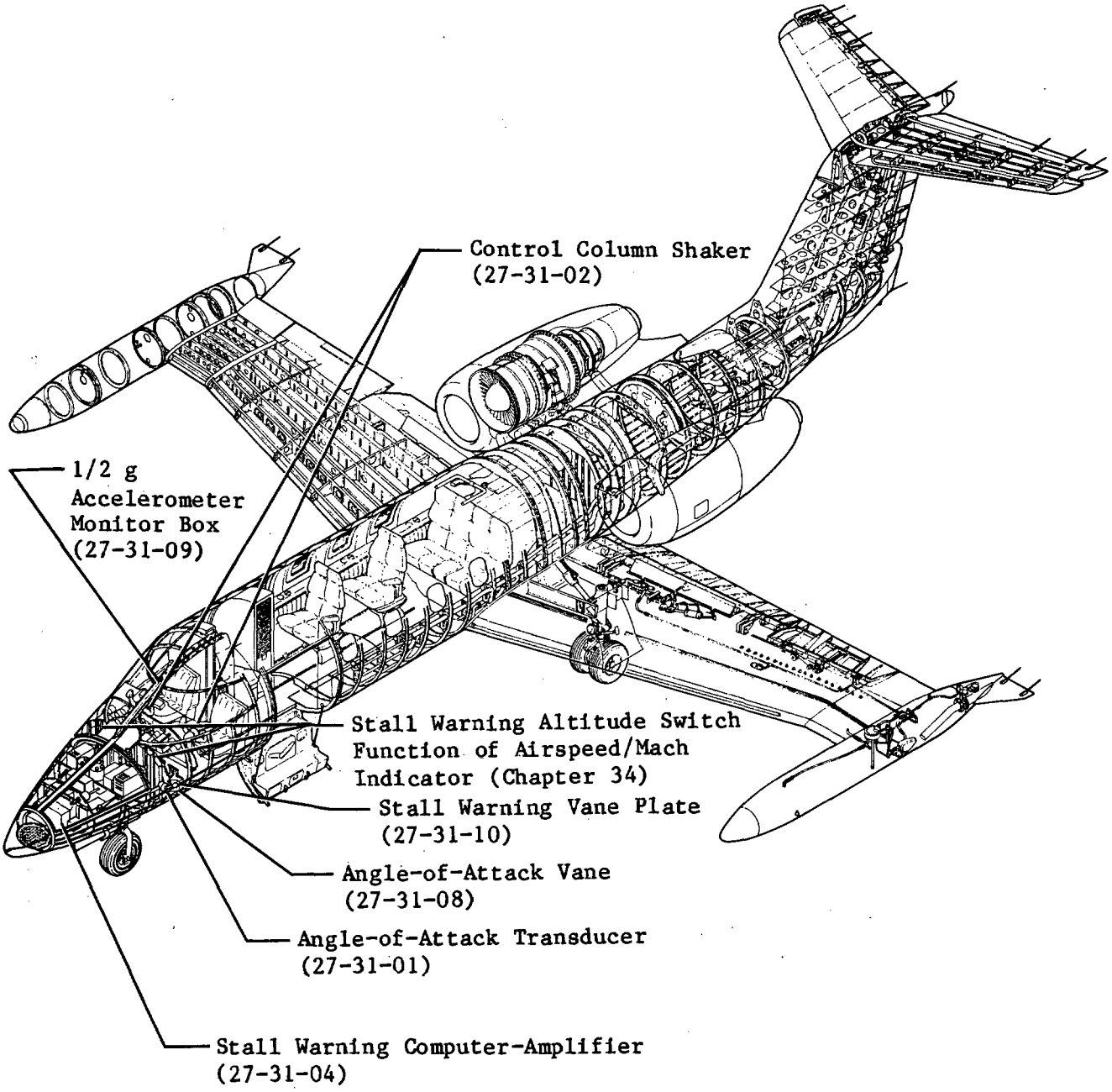




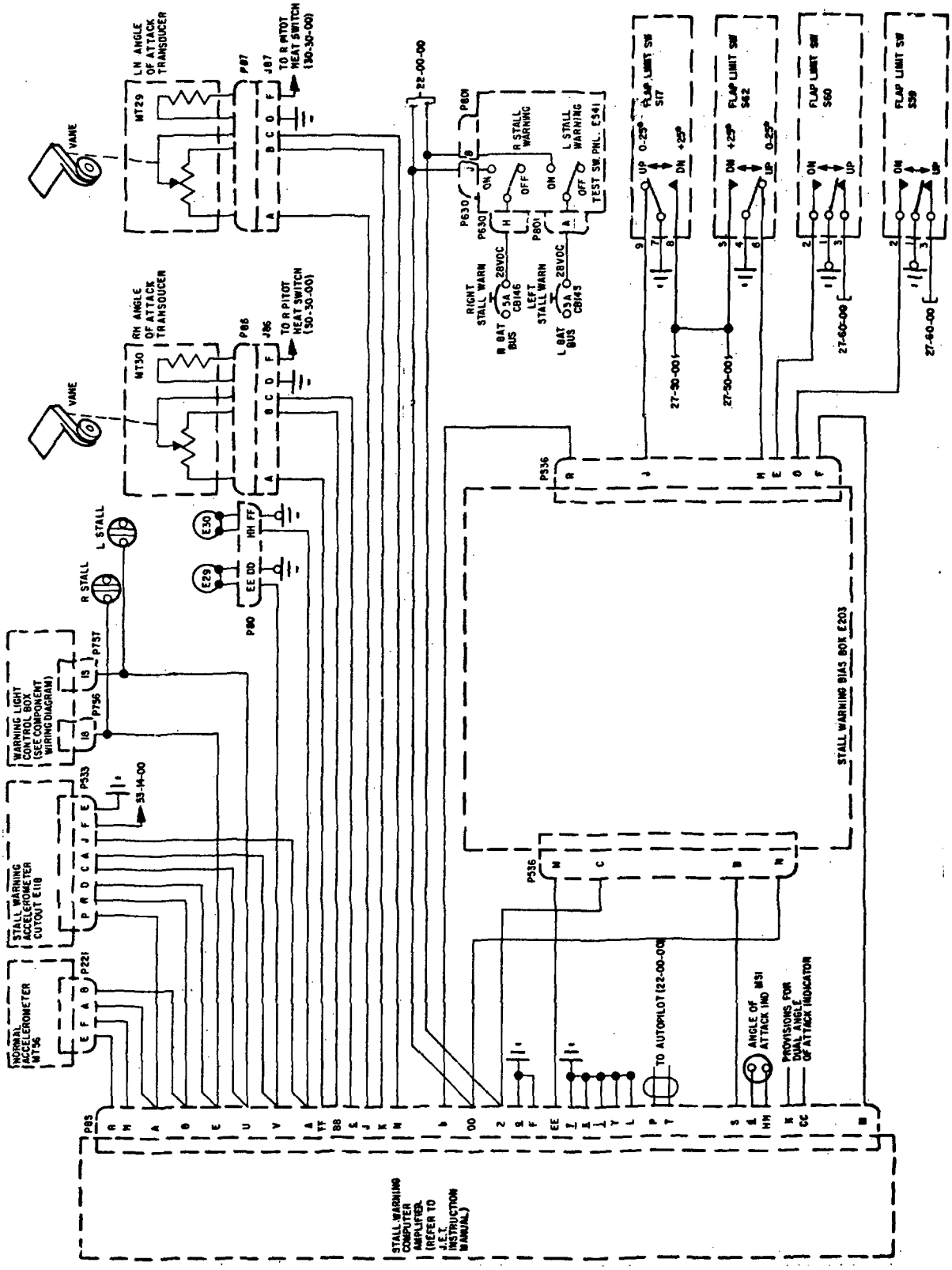
Detail A

9-362A

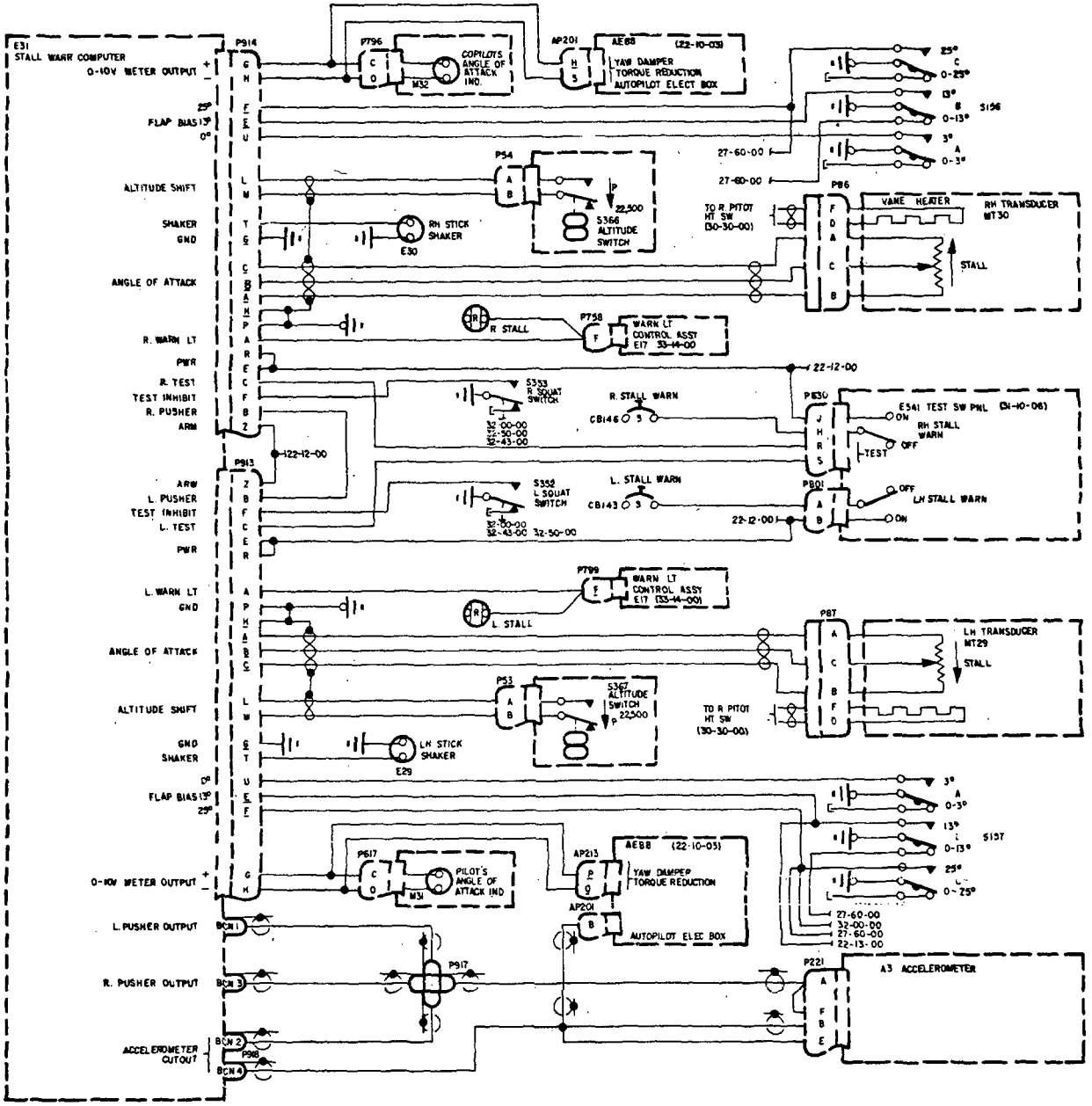
Stall Warning System Component Locator
Figure 1 (Sheet 2 of 3)



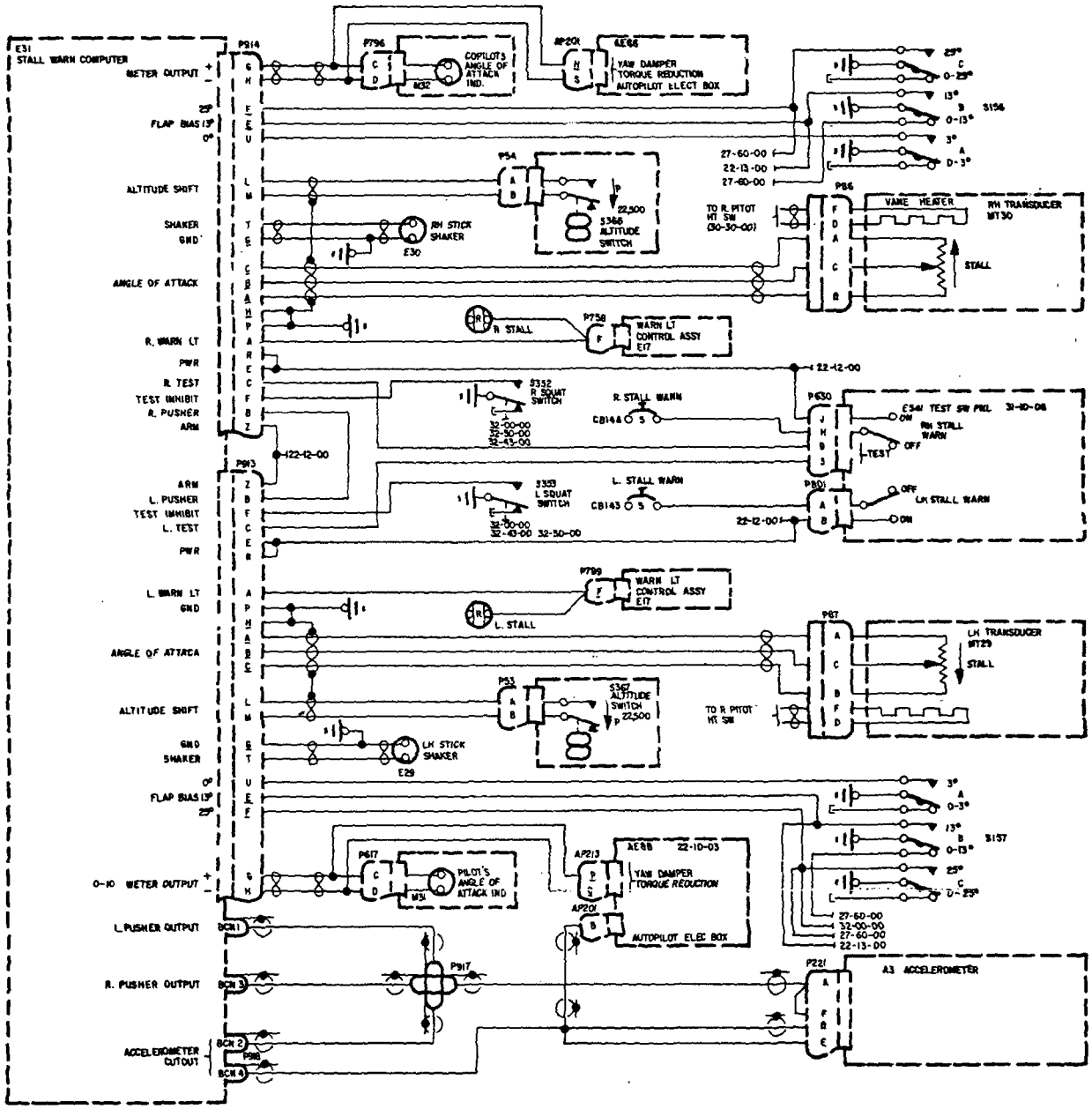
Stall Warning System Component Locator
Figure 1 (Sheet 3 of 3)



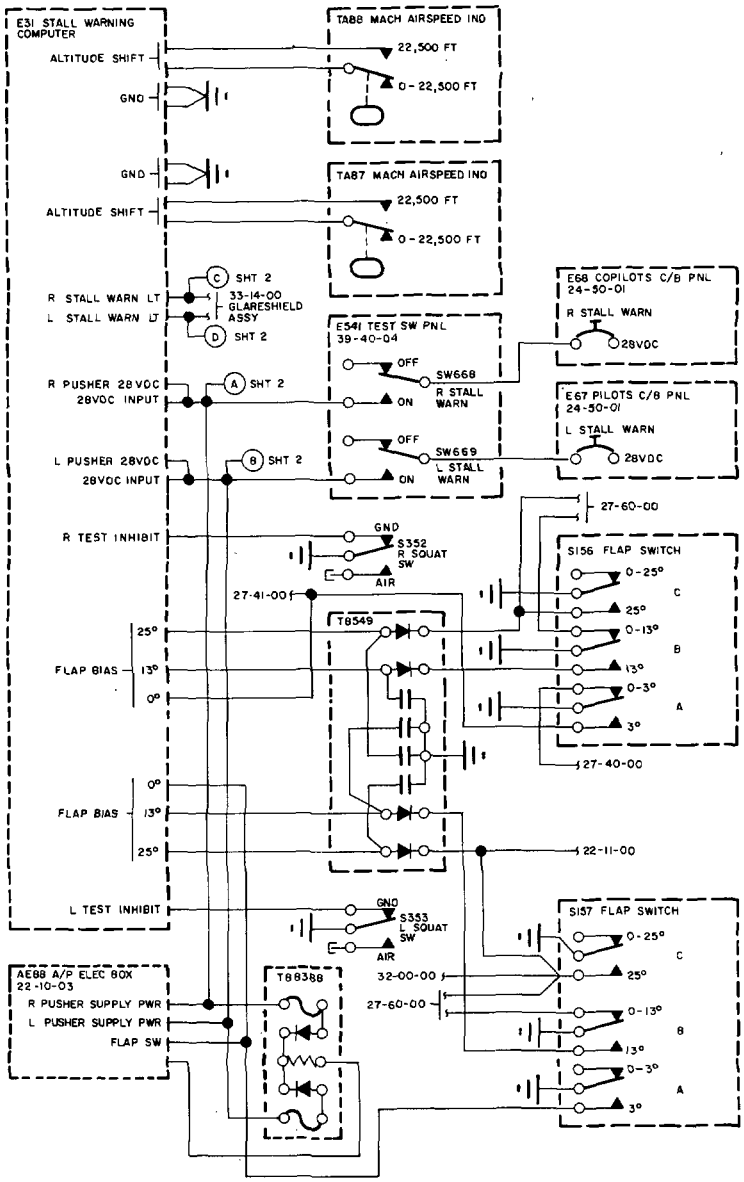
Stall Warning Electrical System Control Schematic
Figure 2



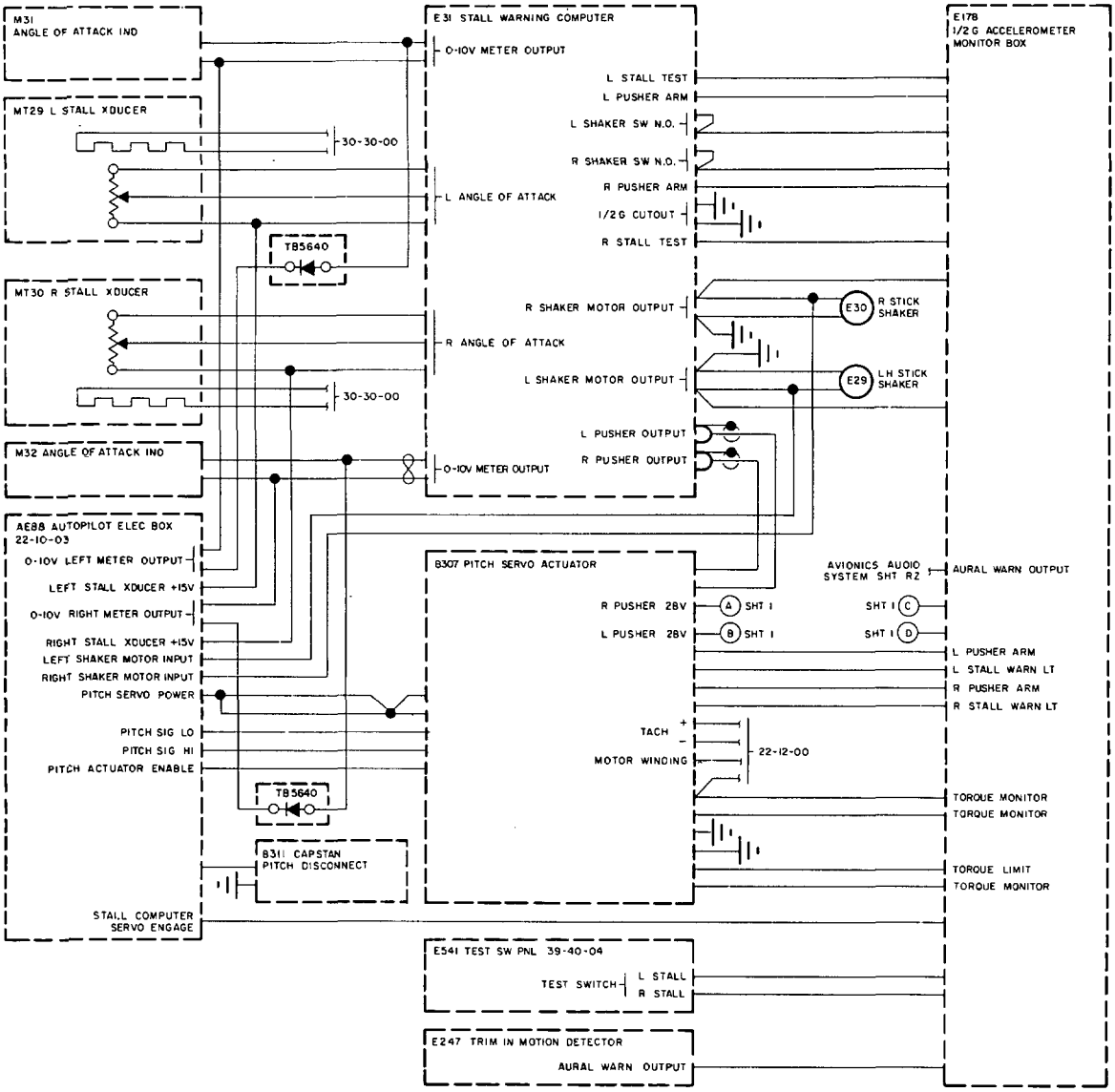
Stall Warning Electrical System Control Schematic
Figure 3



Stall Warning Electrical System Control Schematic
Figure 4

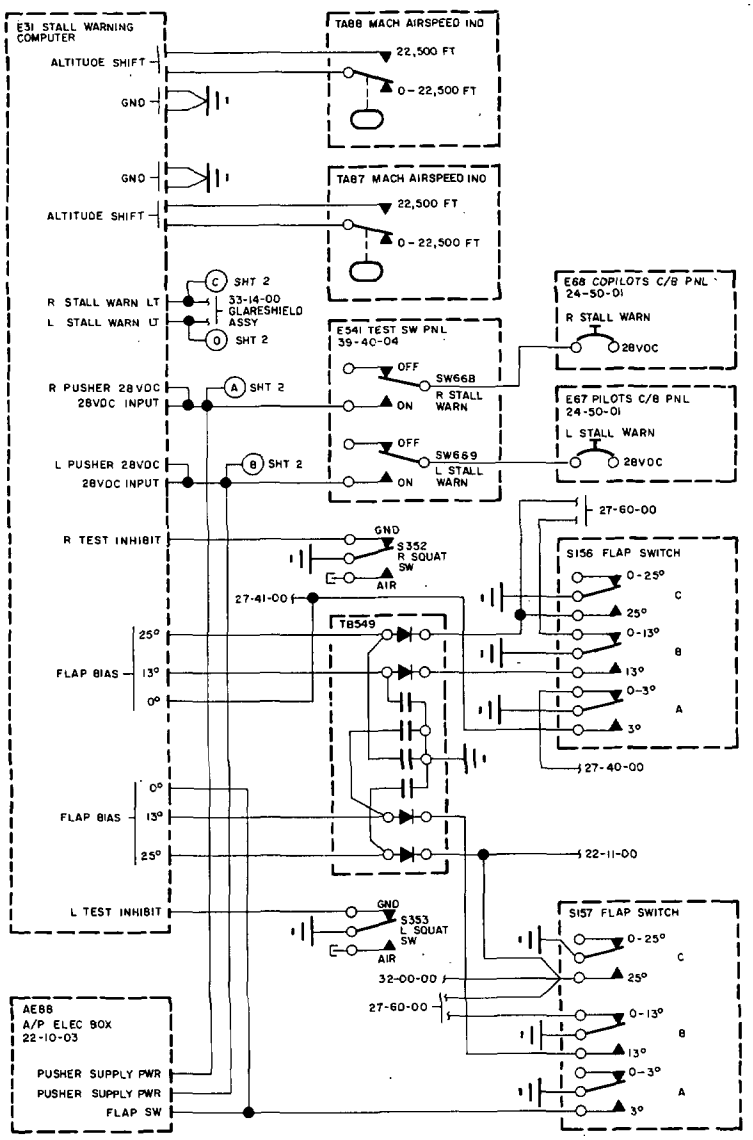


Stall Warning Electrical System Control Schematic
Figure 5 (Sheet 1 of 2)

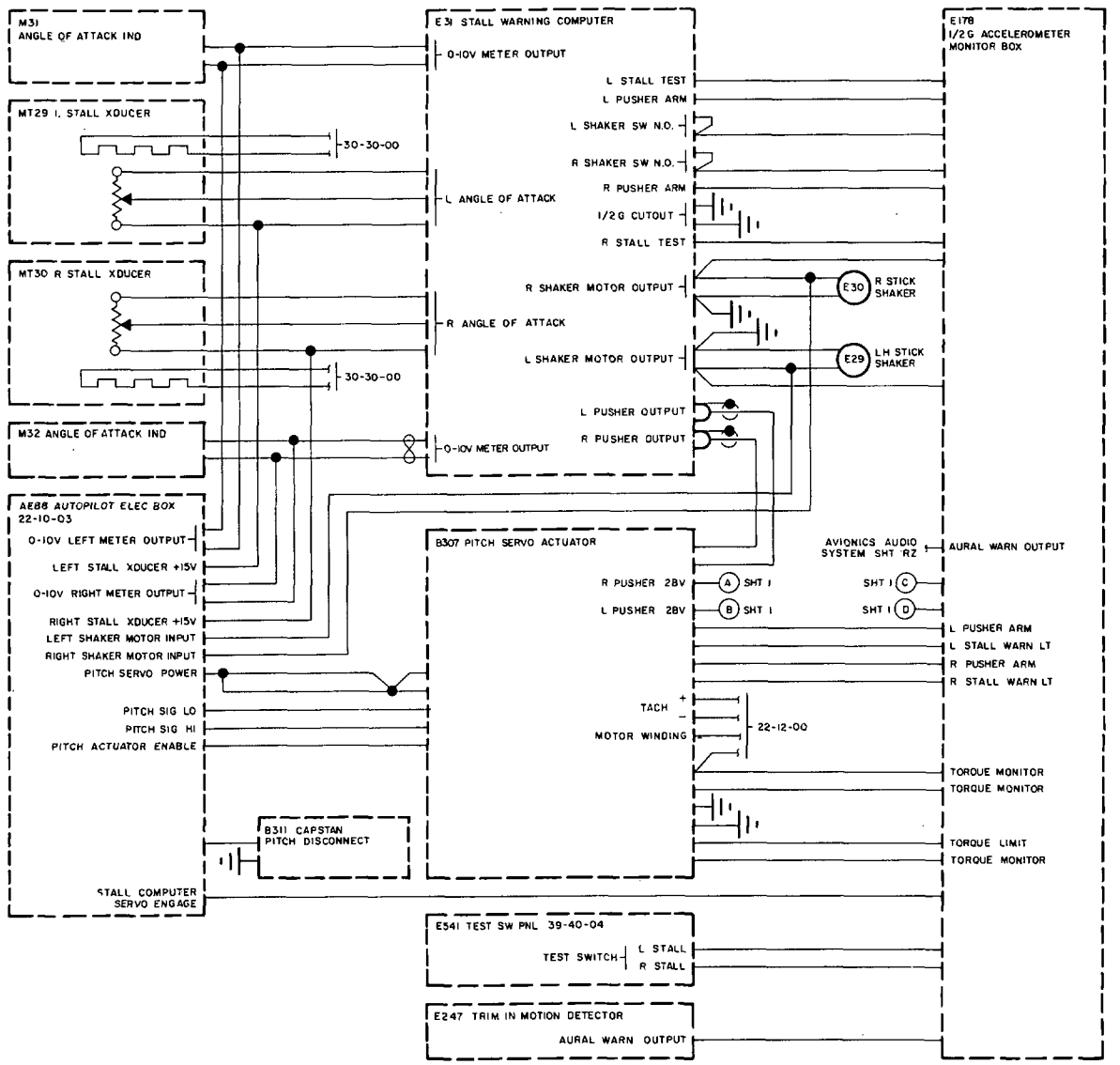


Stall Warning Electrical System Control Schematic
Figure 5 (Sheet 2 of 2)

EFFECTIVITY: Aircraft modified per AAK 83-2,
MM-99 **"Installation of FC-530 Autopilot"**
Disk 535



Stall Warning System Electrical Control Schematic
Figure 6 (Sheet 1 of 2)



Stall Warning System Electrical Control Schematic
Figure 6 (Sheet 2 of 2)

STALL WARNING SYSTEM - TROUBLESHOOTING

1. DESCRIPTION

A. The troubleshooting procedures provided in this section are to be used as an aid to detect and correct possible troubles in the stall warning system.

2. TOOLS AND EQUIPMENT

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Multimeter	Model 260-7	Simpson	Resistance and Voltage Checks
Ohmmeter (Resolution of 1 Ohm When Reading 2500 (± 25) Ohms)		Commercially Available	Resistance Checks

3. TROUBLE SHOOTING

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
1. Stall Warning Stick Shaker or Pusher Malfunction.		
a. Defective Stall Warning Computer.	Perform functional test of Stall Warning System. (Refer to 27-31-00.)	If indicated, replace computer. (Refer to 27-31-04.)
b. Defective angle-of-attack transducer.	Perform functional test of Stall Warning System. (Refer to 27-31-00.)	If test fails, perform Stall Warning System Ground Adjustment. (Refer to 27-31-00.) If problem persists, replace angle-of-attack transducer. (Refer to 27-31-01.)

Stall Warning System Trouble Shooting
Figure 101 (Sheet 1 of 2)

PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
2. Angle-of -attack indicator inoperative.		
c. Defective flap position switch(es).	Perform Gated Flaps System Functional Test. (Refer to 27-51-00.)	If test fails, perform Adjustment of Flap Position Switch. (Refer to 27-51-03.) If problem persists, replace flap position switch(es). (Refer to 27-51-03.)
a. Defective angle-of-attack vane, or transducer.	Visually inspect vane for damage. Check for smooth, free movement	If damaged, replace vane and angle-of-attack transducer. (Refer to 27-31-01.)
	Perform Functional Test of Stall Warning System. (Refer to 27-31-00.)	If indicated, replace angle-of-attack transducer. (Refer to 27-31-01.)
b. Defective Stall Warning Computer.	Perform Functional Test of Stall Warning System. (Refer to 27-31-00.)	If indicated, replace Stall Warning Computer. (Refer to 27-31-04.)
c. Defective Angle-Of-Attack Indicator.	Perform Functional Test of Stall Warning System. (Refer to 27-31-00.)	If indicated, replace Angle-Of-Attack Indicator. (Refer to Chapter 31.)

Stall Warning System Trouble Shooting
Figure 101 (Sheet 2 of 2)

A. Stall Warning System Trouble Shooting Ground Functional Test (*Aircraft 35-067 thru 35-505 and 36-018 thru 36-053 and prior aircraft equipped with Reduced Approach Speed System [AAK 76-4] or Stall Fence [AAK 79-10 or AMK 83-5] and not equipped with AAK 83-2, "Installation of FC-530 Autopilot".)*

WARNING: THE FOLLOWING RESISTANCE CHECKS ARE TO BE PERFORMED WITH POWER REMOVED FROM THE AIRCRAFT. STEPS (7) (g, h, AND i) REQUIRE THE APPLICATION OF HYDRAULIC AND ELECTRICAL POWER TO THE AIRCRAFT. USE EXTREME CAUTION IN PERFORMING THESE STEPS.

NOTE:
- These procedures are provided as a trouble shooting guide only.
- The following check must be performed using an analog meter. A noticeable change in resistance may be experienced if a digital meter is used.

(1) Set Stall Warning Switches to OFF.
(2) Pull L STALL and R STALL circuit breakers.
(3) Ensure that flaps are in full retracted position.
(4) Place aircraft on jacks. (Refer to Chapter 7.)
(5) Disconnect aircraft batteries.
(6) Remove nose compartment access doors. Disconnect electrical connectors from stall warning computer.
(7) Using an ohmmeter, perform the following resistance checks:
(a) Check pins G and H (P913) to aircraft ground. Ohmmeter reading shall be greater than 10k ohms. Record reading on Resistance Check Table, Figure 102.
(b) Check pins G, P, and H (P913) to aircraft ground. Ohmmeter reading shall show continuity. Record reading on Resistance Check Table, Figure 102.
(c) Check pins A, B and C (P913) to aircraft ground. Ohmmeter shall show open circuit. Record reading on Resistance Check Table, Figure 102.

NOTE: On *Aircraft 35-002 thru 35-066; 36-002 thru 36-017*, the LH squat switch controls the RH Stall Warning System and the RH squat switch controls the LH Stall Warning system when performing step (d).

(d) Check pin F (P913) to aircraft ground. With squat switch in air mode (aircraft on jacks) ohmmeter reading shall be greater than 50 ohms. Activate LH squat switch to ground mode. Ohmmeter reading shall be less than 0.5 ohm or less than 1.5 ohms with in-line diode. Record reading on Resistance Check Table, Figure 102.

WARNING: AVOID CONTACT WITH HOT ANGLE-OF-ATTACK VANE.

(e) Check pins A to C (P913). Ohmmeter reading shall be approximately 1.8k ohms. Move LH angle-of-attack vane. Ohmmeter reading shall not change. Record reading on Resistance Check Table, Figure 102.
(f) Check pins B to C (P913). Ohmmeter reading shall be zero to approximately 1.8k ohms and shall vary smoothly with vane movement. Resistance shall decrease as vane is raised. Record reading on Resistance Check Table, Figure 102.
(g) Check pin U (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps slightly, ohmmeter shall read less than 0.5 ohm. Record reading on Resistance Check Table, Figure 102.

(h) Check pin E (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps past 13°. (*Aircraft 35-002 thru 35-144, and 36-002 thru 36-035*), ohmmeter shall read less than 0.5 ohm. (*Aircraft 35-145 and Subsequent, 36-036 and Subsequent*), ohmmeter shall read less than 15 ohms. Record reading on Resistance Check Table, Figure 102.

(i) Check pin F (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps past 25°. (*Aircraft 35-002 thru 35-144, 36-002 thru 36-035*), ohmmeter shall read less than 0.5 ohm. (*Aircraft 35-145 and Subsequent, 36-036 and Subsequent*), ohmmeter shall read less than 15 ohms. Record reading on Resistance Check Table, Figure 102.

(j) Check pin Z (P913) to pin Z (P914). Ohmmeter shall show continuity.

(k) Check pin B (P913) to pin B (P914). Ohmmeter shall show continuity.

(8) Repeat steps (7)(a) thru (7)(k) for RH Stall Warning System using electrical connector P914.

(9) Connect aircraft batteries and set Battery Switches to BAT 1 and BAT 2.

(10) Ensure that Stall Warning switches are set to OFF.

(11) Depress L STALL, R STALL, and WARN LT circuit breakers.

(12) Ensure that flaps are fully retracted.

(13) Using a voltmeter, perform the following voltage checks:

(a) Check pins E and R (P913) to aircraft ground. Voltmeter reading shall be zero volts. Set Stall Warning Switch to L. Voltmeter reading shall be 28 volts. Set L Stall Warning Switch to OFF. Record readings on Voltage Check Table, Figure 103.

(b) Check pin C (P913) to aircraft ground. Voltmeter reading shall be zero volts. Set Stall Warning Switch to L, set System TEST Switch to L STALL and depress pushbutton. Voltmeter reading shall be 28 volts. Set L Stall Warning Switch to OFF. Record reading on Voltage Check Table, Figure 103.

(14) Repeat steps (13)(a) and (13)(b) for the RH Stall Warning System using electrical connector P914.

NOTE: Resolve any wiring discrepancies before proceeding.

(15) Connect electrical connectors to stall warning computer. Ensure that Stall Warning Switches are set to OFF and that Battery Switches are set to BAT 1 and BAT 2. Verify that both L STALL and R STALL annunciators are illuminated (steady) and that both Master Warning annunciators are flashing.

(16) Set Battery Switches to OFF.

(17) Remove aircraft from jacks unless ground calibration must be performed. (Refer to Chapter 7.)

Para.	Probe Pin	Desired Ohmmeter Reading	Actual Ohmmeter Reading	
			LH Stall Sys	RH Stall Sys
(7)(a)	G and H to Aircraft Ground	Greater than 10k ohms		
(7)(b)	G, P, and H to Aircraft Ground	Continuity		
(7)(c)	A, B, and C to Aircraft Ground	Open Circuit		
(7)(d)	F to Aircraft Ground	Squat Switches (Air Mode) Greater than 50 ohms.		
		Squat Switches (Gnd Mode) Less than 0.5 ohm or less than 1.5 ohms with in-line diode.		
(7)(e)	A to C	Approximately 1.8k ohms		
		No change with vane movement.		
(7)(f)	B to C	Zero to approx. 1.8k ohms, varies smoothly with vane position.		
		Resistance decreases as vane is raised.		
(7)(g)	U to Aircraft Ground	Flaps Fully Retracted - Over 50 ohms.		
		Flaps Slightly Lowered - Less than 0.5 ohm.		
(7)(h)	E to Aircraft Ground	Flaps Fully Retracted - Over 50 ohms.		
		Flaps Lowered past 13° - Less than (see 1▷).		
(7)(i)	F to Aircraft Ground	Flaps Fully Retracted - Over 50 ohms.		
		Flaps Lowered past 25° - Less than (see 1▷).		
(7)(j)	Z (P913) to Z (P914)	Continuity		
(7)(k)	B (P913) to B (P914)	Continuity		

1▷ *Effective Aircraft 35-002 thru 35-144, and 36-002 thru 36-035*, less than 0.5 ohm. *Effective Aircraft 35-145 thru 35-505, and 36-036 thru 36-053*, less than 15 ohms.

Resistance Check Table
Figure 102

EFFECTIVITY: 35-067 thru 35-505 and 36-018 thru 36-053 and prior aircraft equipped with Reduced Approach Speed System (AAK 76-4) or Stall Fence (AAK 79-10 or AMK 83-5) and not equipped with AAK 83-2, "Installation of FC-530 Autopilot"

Para.	Probe Pin	Desired Ohmmeter Reading	Actual Ohmmeter Reading	
			LH Stall Sys	RH Stall Sys
(13)(a)	E and R to Aircraft Ground	Stall Warning Switch off - Zero volts.		
		Stall Warning Switch on - 28 volts.		
(13)(b)	C to Aircraft Ground	Zero volts.		
		Test Switch Actuated - 28 volts.		

Voltage Check Table
Figure 103

B. Stall Warning System Trouble Shooting Ground Functional Test (*Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot"*)

WARNING: THE FOLLOWING RESISTANCE CHECKS ARE TO BE PERFORMED WITH POWER REMOVED FROM THE AIRCRAFT. STEPS (7) (g, h, AND i) REQUIRE THE APPLICATION OF HYDRAULIC AND ELECTRICAL POWER TO THE AIRCRAFT. USE EXTREME CAUTION IN PERFORMING THESE STEPS.

NOTE:
- These procedures are provided as a trouble shooting guide only.
- The following check must be performed using an analog VOM. A noticeable change in resistance may be experienced if a digital VOM is used.

(1) Set Stall Warning Switches to OFF.
(2) Pull L STALL and R STALL circuit breakers.
(3) Ensure that flaps are in full retracted position.
(4) Place aircraft on jacks. (Refer to Chapter 7.)
(5) Disconnect aircraft batteries.
(6) Remove nose compartment access doors. Disconnect electrical connectors from stall warning computer.
(7) Using an ohmmeter, perform the following resistance checks:
  (a) Check pins G to H (P913). Ohmmeter reading shall be greater than 7.5k ohms. Record reading on Resistance Check Table, figure 104.
  (b) Check pins G, P, B, and D (P913) to aircraft ground. Ohmmeter reading shall show continuity. Record reading on Resistance Check Table, figure 104.
  (c) Check pins M and L (P913) to aircraft ground. Ohmmeter shall show open circuit. Record reading on Resistance Check Table, figure 104.
  (d) Check pins A to C (P913). Ohmmeter reading shall be approximately 2k ohms. Move LH angle-of-attack vane. Ohmmeter reading shall not change. Record reading on Resistance Check Table, figure 104.
  (e) Check pins B to C (P913). Ohmmeter reading shall be zero to approximately 2k ohms and shall vary smoothly with vane movement. Resistance shall decrease as vane is raised. Record reading on Resistance Check Table, figure 104.
  (f) Check pin U (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps slightly, ohmmeter shall read less than 15 ohm. Record reading on Resistance Check Table, figure 104.
  (g) Check pin E (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps past 13°, ohmmeter shall read less than 15 ohm. Record reading on Resistance Check Table, figure 104.
  (h) Check pin F (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps past 25°, ohmmeter shall read less than 15 ohm. Record reading on Resistance Check Table, figure 104.
  (i) Check pins C, B, and A to aircraft ground. Ohmmeter reading shall be greater than 7.5k ohms. Record reading on Resistance Check Table, figure 104.
(8) Repeat steps 7(a) thru 7(i) for RH Stall Warning System Using electrical connector P914.
(9) Connect aircraft batteries and set battery Switches on.
(10) Ensure that Stall Warning Switches are set to OFF.
(11) Depress L STALL, R STALL, and WARN LT circuit breakers.
(12) Ensure that flaps are fully retracted.

(13) Using a voltmeter, perform the following voltage checks:
  (a) Check pins E and R (P913) to aircraft ground. Voltmeter reading shall be zero volts. Set stall warning switch to L. Voltmeter reading shall be zero volts. Set stall warning switch to L. Voltmeter reading shall be 28 volts. Set battery switches to OFF. Note that there is no effect. Pull L STALL circuit breaker. Voltmeter reading shall be zero volts. Depress L STALL circuit breaker. Set Battery Switches to BAT 1 and BAT 2. Record readings on Voltage Check Table, figure 105.
  (b) Check pin C (P913) to aircraft ground. Voltmeter reading shall be zero volts. Set system TEST Switch to L STALL and depress pushbutton. Voltmeter reading shall be 28 volts. A short aural warning shall sound. Set L Stall Warning Switch to OFF. Record reading on Voltage Check Table, figure 105.
  (c) Jumper pin A (P913) to ground. The L STALL warning annunciator shall illuminate and the Master Warning annunciators shall flash. Remove Jumper. L STALL and Master Warning annunciators shall extinguish.

(14) Repeat steps 13(a) thru (c) for the RH Stall Warning System using electrical connector P914, R STALL circuit Breaker, and R Stall Warning Switch.

NOTE: Resolve any wiring discrepancies before proceeding.

(15) Connect electrical connectors to stall warning computer. Ensure that Stall Warning Switches are set to OFF and that Battery Switches are set to BAT 1 and BAT 2. Verify that both L STALL and R STALL annunciators are flashing.
(16) Set Battery Switches to OFF.
(17) Remove aircraft from jacks unless ground calibration must be performed. (Refer to Chapter 7.)

Para.	Probe Pin	Desired Ohmmeter Reading	Actual Ohmmeter Reading	
			LH Stall Sys	RH Stall Sys
(7)(a)	G to H	Greater than 7.5k ohms		
(7)(b)	G, P, B, and D to Aircraft Ground	Continuity		
(7)(c)	M and L to Aircraft Ground	Open Circuit		
(7)(d)	A to C	Approximately 2k ohms		
		No change with vane movement		
(7)(e)	B to C	Zero to approx. 2k ohms, varies smoothly with vane position.		
		Resistance decreases as vane is raised.		
(7)(f)	U to Aircraft Ground	Flaps Fully Retracted - Over 50 ohms.		
		Flaps Slightly Lowered - Less than 15 ohm.		
(7)(g)	E to Aircraft Ground ▷	Flaps Fully Retracted - Over 50 ohms.		
		Flaps Lowered past 13° - Less than 15 ohm.		
(7)(h)	F to Aircraft Ground ▷	Flaps Fully Retracted - Over 50 ohms.		
		Flaps Lowered past 25° - Less than 15 ohm.		
(7)(i)	C, B, and A to Aircraft Ground	Greater than 7.5k ohms.		

▷ Read resistance with analog volt-ohmmeter. Do not use a digital ohmmeter.

Resistance Check Table
Figure 104

Para.	Probe Pin	Desired Ohmmeter Reading	Actual Ohmmeter Reading	
			LH Stall Sys	RH Stall Sys
(13)(a)	E and R to Aircraft Ground	Stall Warning Switch off - Zero volts		
		Stall Warning Switch on - 28 volts		
		Battery Switches off - 28 volts		
		L STALL circuit breaker - Zero volts		
(13)(b)	C to Aircraft Ground	Zero volts		
		Test Switch Actuated - 28 volts		
(13)(c)	Jumper A to Aircraft Ground	L STALL and Master Warning annunciators flash		

Voltage Check Table
Figure 105

STALL WARNING - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Hydraulic Power Cart	Model 05-3005-1100	Tronair Inc. Everett, WA	Hydraulic power.
Ground Power Cart	6T28-400CL	Hobart	Electrical power.
Protractor		Commercially Available	Check angles.
Adhesive	Pro Seal 727	Products Research & Chemical Co. Glendale, CA	Seal transducer opening.
Sealant	Pro-Seal 890B	Products Research & Chemical Co. Glendale, CA	Seal access hole.
Pitot-Static Tester	1811H-466	Barfield Instrument Co.	Check system.
Patch Cable*	15-1013	Learjet Inc. Wichita, KS	Check system.
Inclinometer	T3500002-600	Learjet Inc. Wichita, KS	Calibration procedure.
Calibration Box (With Cables)	5471100-7	Learjet Inc. Wichita, KS	Calibration procedure.
Masking Tape		Commercially Available	Protect aircraft surface.
Mylar Template	5471102-3 (LH)** 5471102-4 (RH)** 5471102-7 (LH)*** 5471102-8 (RH)***	Learjet Inc. Wichita, KS	Calibration procedure.

* Refer to Paragraph 2 for effectivity.

** *Aircraft 35-279 thru 35-505, and 36-045 thru 36-053 and prior aircraft equipped with stall fence (AAK 79-10 or AMK 83-5).*

*** *Aircraft 35-408, 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot."*

TEMPORARY REVISION NO. 27-62

MANUALS AFFECTED:	Learjet 35/35A/36/36A Maintenance Manual.
FILING INSTRUCTIONS:	Insert adjacent to 27-31-00, Page 202, dated Jun 25/93 and retain until further notice.
REASON:	Add NOTE to include aircraft equipped with Rosemount pitot-static probes and the FC-200 autopilot.
INSTRUCTIONS:	Add (6) to NOTE as follows:

STALL WARNING - MAINTENANCE PRACTICES

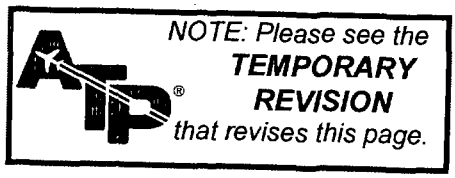
1. **Tools and Equipment** (No change to existing information)

2. **Adjustment/Test**

NOTE: (6) *On aircraft 35-002 thru 35-676 and 36-002 thru 36-063 modified per SB 35/36-27-43*, refer to Supplemental Maintenance Manual M100165ICT-SMM-3 for stall warning system maintenance practices. This Maintenance Manual covers aircraft with the Rosemount pitot-static probes and the FC-200 autopilot.

2. Adjustment/Test

NOTE:
- The following aircraft serial numbers and adjustment/test paragraph references are provided to aid maintenance personnel.
  (1) *Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System (AAK 76-4), Stall Fence (AAK 79-10 or AMK 83-5), or AAK 83-2, "Installation of FC-530 Autopilot,"* must comply with paragraph 2.A. If the aircraft stall warning system complies with paragraph 2.A., then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 2.A., then paragraphs 2.B. and 2.C. must be complied with.
  (2) *Aircraft 35-067 thru 35-278 and 36-018 thru 36-044 and prior aircraft equipped with Reduced Approach Speed System (AAK 76-4) but not equipped with Stall Fence (AAK 79-10 or AMK 83-5), or AAK 83-2, "Installation of FC-530 Autopilot,"* must comply with paragraph 2.D. If the aircraft stall warning system complies with paragraph 2.D., then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 2.D., then paragraphs 2.E. and 2.F. must be complied with.
  (3) *Aircraft 35-279 thru 35-505 and 36-045 thru 36-053 and prior aircraft equipped with Stall Fence (AAK 79-10 or AMK 83-5), and NOT modified by AAK 83-2, "Installation of FC-530 Autopilot,"* must comply with paragraph 2.D. If the aircraft stall warning system complies with paragraph 2.D., then aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 2.D., then one of the following must be complied with:
    (a) Paragraphs 2.E. and 2.G.
    (b) Paragraph 2.H.
  (4) *Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot,"* must comply with paragraph 2.D. If the aircraft stall warning system complies with paragraph 2.D., then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 2.D., then one of the following must be complied with:
    (a) Paragraphs 2.E. and 2.G.
    (b) Paragraph 2.H.
  (5) *Aircraft 35-002 and Subsequent and 36-002 and Subsequent requiring an angle-of-attack vane replacement, due to a defective heater*, must comply with paragraphs 2.A. or 2.D. as applicable following the part replacement. If the aircraft stall warning system complies with paragraphs 2.A. or 2.D., as applicable, then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraphs 2.A. or 2.D., as applicable, then paragraphs 2.E., 2.F., 2.G., or 2.H., as applicable, must be complied with.



NOTE:

- Due to the possible effects on stall characteristic, whenever maintenance as noted in items (1) thru (4) below is performed, the aircraft must be flight tested to verify aerodynamic acceptability and stall speeds. This flight test must be conducted by a pilot approved by Learjet Corporation for stall test flights. The stall warning system must be checked in accordance with the flight or ground calibration procedures in 27-31-00 of this maintenance manual when maintenance as noted in items (1) thru (5) is performed.

  (1) When maintenance requiring removal and installation, repair, or installation of a new wing leading edge is performed. This includes loosening and retightening any of the leading edge screws.
  (2) When a new tip tank strake is installed. *Applies to 35-067 and Subsequent and 36-018 and Subsequent and prior aircraft equipped with Reduced Approach Speed and/or equipped with Stall Fence (AAK 79-10 or AMK 83-5).*
  (3) When a new tip tank is installed. *Applies to 35-067 and Subsequent and 36-018 and Subsequent and prior aircraft equipped with Reduced Approach Speed and/or equipped with Stall Fence (AAK 79-10 or AMK 83-5).*
  (4) When maintenance requiring repair to, or installation of, new stall strips. *Applies to 35-279 and Subsequent and 36-045 and Subsequent and prior aircraft equipped with Stall Fence (AAK 79-10 or AMK 83-5).*
  (5) When an angle-of-attack transducer, stall warning computer, or indicator is repaired, replaced, or calibrated.

- On *Aircraft 35-067 and Subsequent and 36-018 and Subsequent and prior aircraft equipped with Reduced Approach Speed*, replacement of the angle-of-attack vane is permitted in the event of a failed heater. Physical damage to the vane requires total transducer replacement because of possible damage to the transducer shaft, bearing, and potentiometer. Vane replacement requires functional test to verify that shaker and pusher actuation occurs within the prescribed tolerance of the vane plate marks.

- If angle-of-attack indicator replacement does not require an IND or Rx potentiometer adjustment and a quick check shows indicator to be at intersection of yellow/green for shaker and at intersection of red/ yellow for pusher then the aircraft does not require flight test/ adjustment or ground calibration adjustment.

- Perform functional test of stall warning system in accordance with the current inspection interval specified in Chapter 5.

- A flashing stall warning annunciator indicates proper system operation. A steady stall warning annunciator indicates system malfunction.

- Angle-of-attack vanes must be moved slowly to give correct indication. The rate of vane movement is used in stall computation. Failure to move vanes slowly will cause angle-of-attack indicator needle movement to be jumpy.

- Components of stall warning system that do not meet functional test requirements shall be replaced or recalibrated.

- Pusher activation shall occur when the vane is positioned at the appropriate vane plate mark +0, -1/32 inch (+ 0, - 0.794 millimeters). Shaker activation shall occur when the vane is positioned at the appropriate vane plate mark ± 1/16 inch (± 1.59 millimeter). A positive (+) error means the vane is above the appropriate vane plate mark. A negative (-) error means the vane is below the appropriate vane plate mark.

- On *Aircraft 35-279 thru 35-505, 36-045 thru 36-053 and prior aircraft equipped with Stall Fence (AAK 79-10 or AMK 83-5)*, vane plate marks can be verified or corrected using mylar templates, P/N 5471102-3 (left) and P/N 5471102-4 (right), available from Learjet Corporation.

A. Functional Test of Stall Warning System (*Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System [AAK 76-4], Stall Fence [AAK 79-10 or AMK 83-5], or modified by AAK 83-2, "Installation of FC-530 Autopilot."*) (See figure 201.)

(1) Set Battery Switches on.

WARNING: AVOID CONTACT WITH HOT VANES.

(2) Set Pitot Heat Switches to L and R and verify that vanes are heated. Set Pitot Heat Switches to OFF.

(3) Set Left Stall Warning Switch to ON and ensure that the left stall warning light extinguishes and right stall warning light remains illuminated. Allow 15-minute warmup of stall warning computer.

(4) Hold control wheel and observe the stall warning lights while second person aids in performing the following:

(a) Extend flaps to full down (40°).

(b) Lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line labeled •••. The shaker shall actuate and the left stall warning light will flash.

(c) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on the upper line labeled •••. The pusher shall actuate and the left stall warning light will continue to flash and shaker will continue to shake.

(d) Retract flaps (to 20°).

(e) Lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line labeled ••. The shaker shall actuate and the left stall warning light will flash.

(f) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line labeled ••. The pusher shall actuate and the left stall warning light will continue to flash and shaker will continue to shake.

(g) Completely retract flaps.

(h) Lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line labeled •. The shaker shall actuate and the left stall warning light will flash.

(i) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on the upper line labeled •. The pusher shall actuate and the left stall warning light will continue to flash and shaker will continue to shake.

(j) Repeat steps (3) thru (4)(i) using the right angle-of-attack vane and Right Stall Warning Switch.

(k) Set Battery Switches and Stall Warning Switches off.

(l) Check upper travel limit of angle-of-attack vanes as shown in figure 201.

(m) If the preceding stall system functional test fails, then the stall warning system ground adjustment (paragraph 2.B.) and the stall warning system flight test/adjustment (paragraph 2.C.) must be performed.

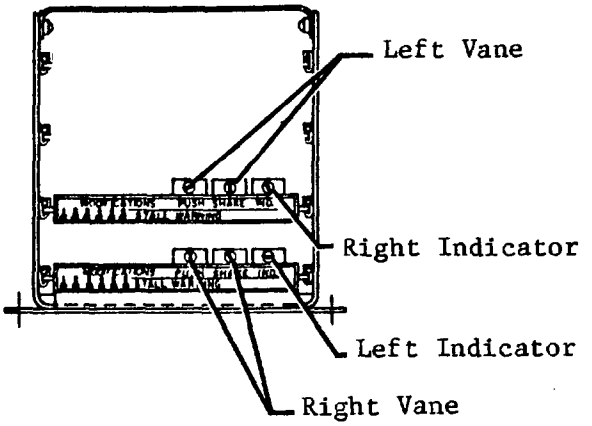
B. Stall Warning System Ground Adjustment (*Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System (AAK 76-4), Stall Fence (AAK 79-10 or AMK 83-5), or modified by AAK 83-2, "Installation of FC-530 Autopilot."* (See figure 202.)

NOTE:
- The Ground Adjustment of this paragraph is a preliminary calibration to the required flight adjustment.
- The pusher-shaker systems should be adjusted only by completely qualified personnel, preferably at a factory-authorized service center. It will be necessary to fabricate patch cables in order for the stall warning computer and the stall warning bias box to be accessible in the cabin during the flight required by paragraph 2.C.

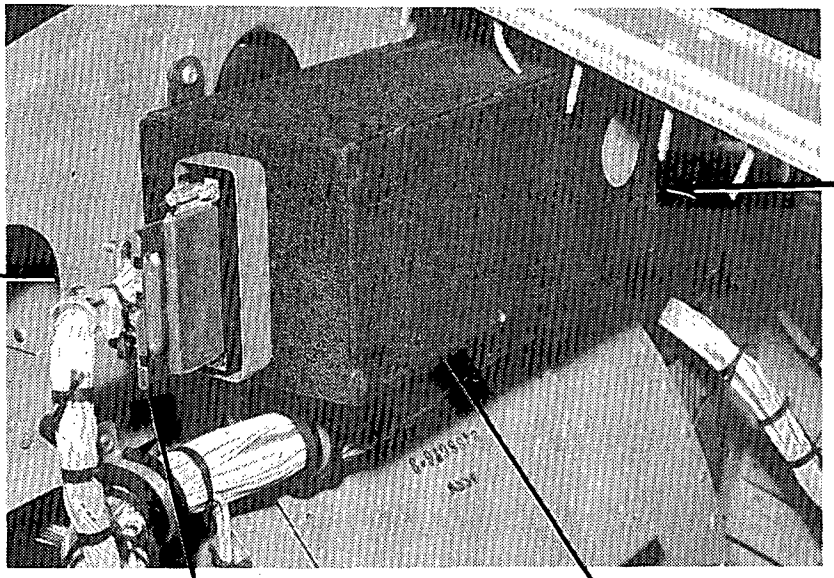
(1) Remove nose compartment access doors.
(2) Connect auxiliary electrical and hydraulic power units to aircraft.
(3) Potentiometers R23 through R25, R32, R272 thru R275 are externally adjustable on the stall warning bias box. On *Aircraft 35-002 thru 35-061 and 36-002 thru 36-017 not modified per AMK76-8, "Angle-of-Attack Indication Improvement,"* adjust R23, R24, R25, and R32 fully clockwise. On *Aircraft 35-062 thru 35-066 and prior aircraft modified per AMK76-8, "Angle-of-Attack Indication Improvement,"* adjust R23, R24, R25, R273, and R275 fully clockwise and R272 and R274 to mid-range.
(4) Remove cover from stall warning computer (end opposite plug). Using a small screwdriver, turn the left and right PUSH, SHAKE, and IND potentiometers fully clockwise until they click.
(5) Set Battery Switches and Left Stall Warning Switch to ON. Allow 15-minute warmup of stall warning computer.
(6) With a service person in the cockpit, holding control wheel and observing stall warning lights and angle-of-attack indicator, perform the following:
  (a) Extend flaps to full down (40°).
  (b) Lift left angle-of-attack vane until aft lower edge of vane is centered on lower line labeled •••. Turn left SHAKE potentiometer counterclockwise until shaker actuates and the left stall warning light flashes.
  (c) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line labeled •••. Turn left PUSH potentiometer counterclockwise until pusher actuates and left stall warning light flashes. On *Aircraft 35-002 thru 35-061 and 36-002 thru 36-017 not modified per AMK76-8, "Angle-of-Attack Indication Improvement,"* turn the right IND potentiometer counterclockwise until the angle-of-attack indicator needle is at the intersection of the amber and red bands. Move the angle-of-attack vane to the shaker position; the angle-of-attack indicator needle will be within the proximity of the intersection of the green and amber bands. On *Aircraft 35-062 thru 35-066 and prior aircraft modified per AMK76-8, "Angle-of-Attack Indication Improvement,"* adjust R275 potentiometer counterclockwise until the angle-of-attack indicator needle is at the intersection of the amber and red bands. Move the angle-of-attack vane to the shaker position; the angle-of-attack indicator needle will be within the proximity of the intersection of the green and amber bands. Potentiometer R274 may be adjusted to expand the range.

    NOTE: Due to the interaction of the pusher and shaker points, repeat steps (6)(b) and (6)(c) to stabilize each adjustment.

  (d) Retract flaps to 20°. Check with a protractor. Do not rely on position indicator.
  (e) Lift left angle-of-attack vane until aft lower edge of vane is centered on lower line labeled ••. Turn potentiometer R25 counterclockwise until shaker actuates and left stall warning light flashes.



Detail A



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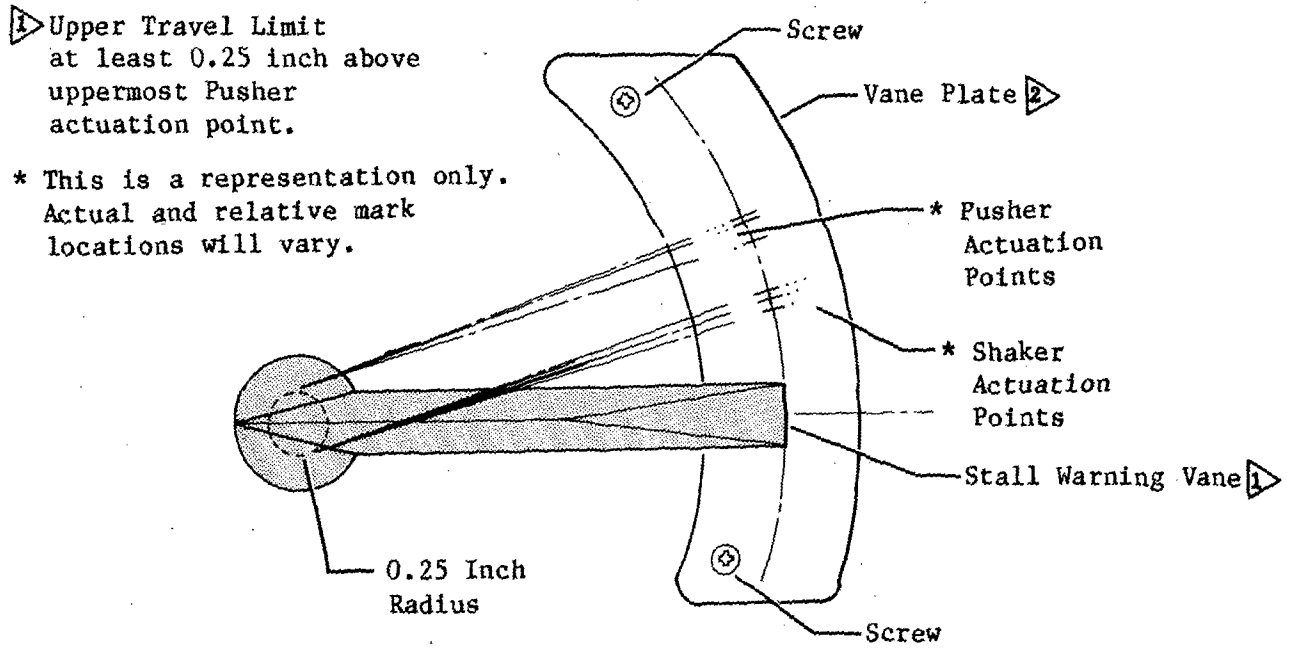
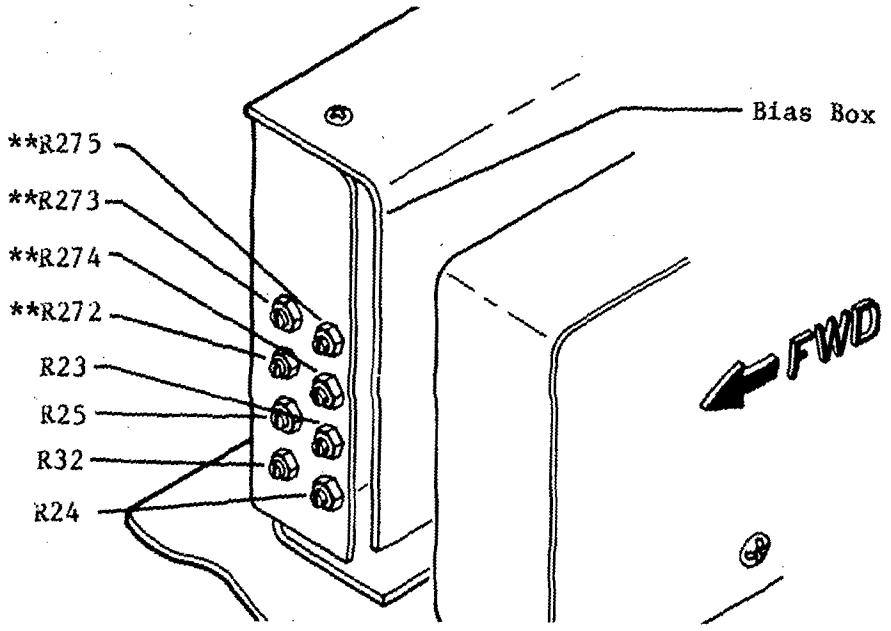
Stall Warning Adjustment and Test
Figure 201 (Sheet 1 of 2)

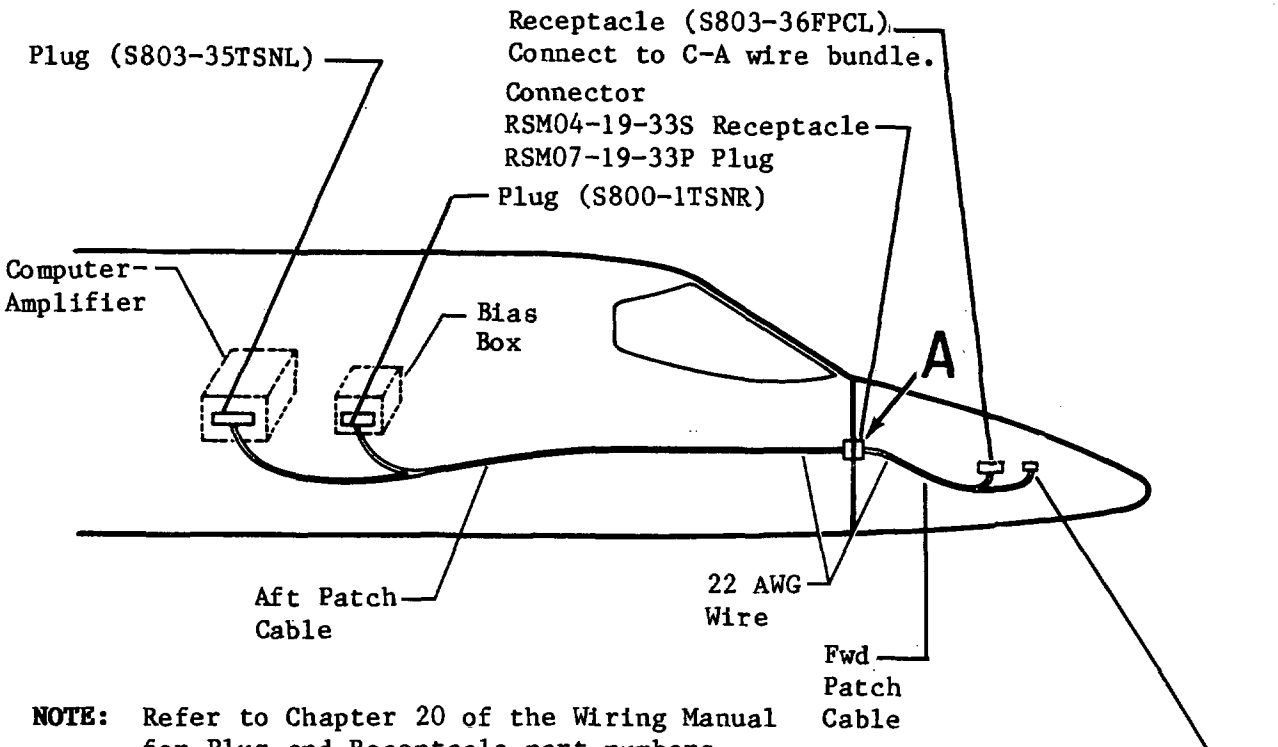


2 Vane Plate is bonded to aircraft skin with EC-2216 adhesive, mfd. by 3M Co.

** Potentiometers effective for Aircraft 35-062 thru 35-066 and prior Aircraft modified per AMK76-8, "Angle-of-Attack Indication Improvement."

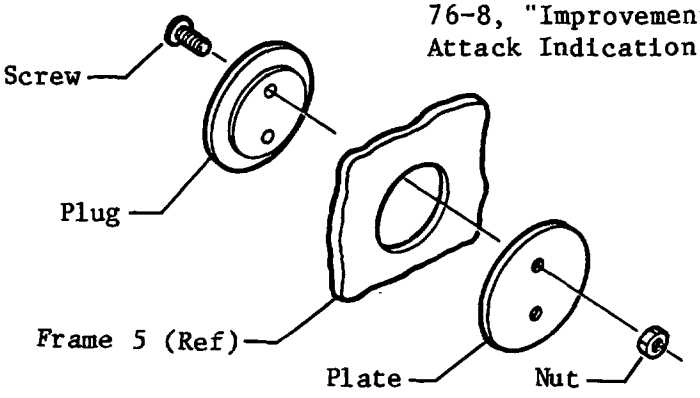


Stall Warning Adjustment and Test
Figure 201 (Sheet 2 of 2)



NOTE: Refer to Chapter 20 of the Wiring Manual for Plug and Receptacle part numbers.

Receptacle (S800-2FPO, Aircraft 35-002 thru 35-008; 36-002 thru 36-004) (S800-6FPO, Aircraft 35-009 thru 35-066; 36-005 thru 36-017 and Aircraft Modified per AMK 76-8, "Improvement of Angle-of-Attack Indication.")



Detail A

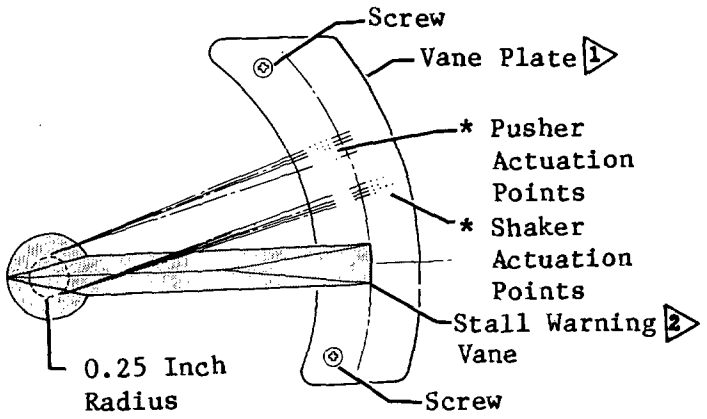
Fabrication and Use of Patch Cables
Figure 202

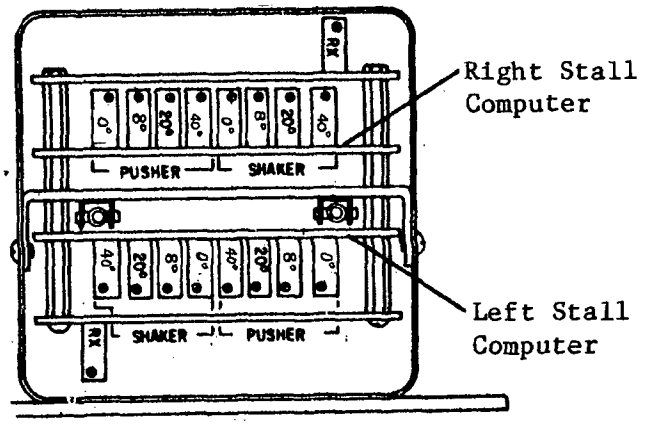
(f) Continue to lift angle-of-attack vane until aft upper edge is centered on upper line labeled ••. Pusher shall actuate and left stall warning light will flash.
(g) Completely retract flaps.
(h) Lift left angle-of-attack vane until aft lower edge of vane is centered on lower line labeled •. Turn potentiometer R23 counterclockwise until shaker actuates and left stall warning light flashes.
(i) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line labeled •. Pusher shall actuate and left stall warning light will flash.
(j) Set Left Stall Warning Switch OFF and Right Stall Warning Switch ON.
(k) Extend flaps to full down (40°).
(l) Lift right angle-of-attack vane up until aft lower edge of vane is centered on lower line labeled •••. Turn right SHAKE potentiometer counterclockwise until shaker actuates and right stall warning light flashes.
(m) Continue to lift the right angle-of-attack vane up until aft upper edge of vane is centered on upper line labeled •••. Turn right PUSH potentiometer counterclockwise until the pusher actuates and right stall warning light flashes. On *Aircraft 35-002 thru 35-061 and 36-002 thru 36-017 not modified per AMK76-8, "Angle-of-Attack Indication Improvement,"* turn the left IND potentiometer counter-clockwise until the angle-of-attack indicator needle is at the intersection of the amber and red bands. Move the angle-of-attack vane to the shaker position. The angle-of-attack indicator needle will be within the proximity of the intersection of the green and amber bands. On *Aircraft 35-062 thru 35-066 and prior aircraft modified per AMK76-8, Angle-of-Attack Indication Improvement,"* adjust potentiometer R273 counterclockwise until the angle-of-attack indicator needle is at the intersection of the amber and red bands. Move vane to the shaker position and check that the angle-of-attack indicator needle is approximately at the intersection of the green and amber bands. Potentiometer R272 may be adjusted to expand the range.

NOTE: Repeat steps (l) and (m) to stabilize each adjustment.

(n) Retract flaps to 20°. Check with a protractor. Do not rely on position indicator.
(o) Lift right angle-of-attack vane until aft lower edge is centered on lower line labeled ••. Turn potentiometer R24 counterclockwise until shaker actuates and right stall warning light flashes.
(p) Continue to lift angle-of-attack vane until aft upper edge is centered on upper line labeled ••. Pusher shall actuate and right stall warning light will flash.
(q) Retract flaps to full up.
(r) Lift right angle-of-attack vane until aft lower edge of vane is centered on lower line labeled •. Turn potentiometer R32 counterclockwise until shaker actuates and right stall warning light flashes.
(s) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line labeled •. Pusher shall actuate and right stall warning light will flash.
(t) Check both vanes for upper travel limit as shown in figure 201. If adjustment is necessary, remove transducer cover and rotate adjusting screw until the upper edge (for pusher) of vane stops at least 0.25 inch (6.35 millimeter) above highest mark on plate. Install transducer cover and safety. Apply Pro Seal 727 around pigtail at transducer opening, left transducer only.
(u) Set Battery Switches and Right Stall Warning Switch to OFF.

* This is a representation only. Actual and relative mark locations will vary.

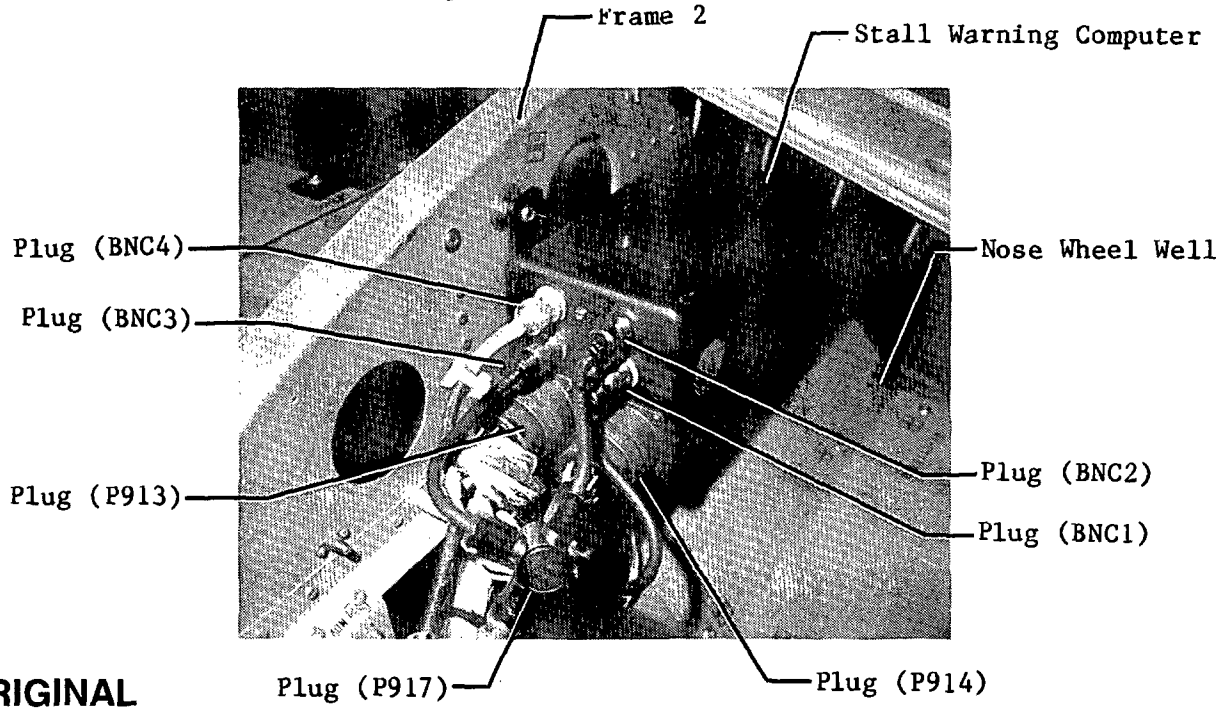




END OPPOSITE PLUG SHOWN
COVER REMOVED

1 Vane plate is bonded to aircraft skin with EC-2216 adhesive, mfd. by 3M Co.

2 Upper travel limits at least 0.25 inch above uppermost pusher activation point.



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Stall Warning System Adjustment and Test
Figure 203

C. Stall Warning System Flight Test/Adjustment (*Aircraft 35-002 thru 35-066 and 36-002 thru 36-017 not equipped with Reduced Approach Speed System (AAK 76-4), Stall Fence (AAK 79-10 or AMK 83-5), or modified by AAK 83-2, "Installation of FC-530 Autopilot."*)

WARNING:
- **STALL WARNING SYSTEM FLIGHT TESTING MUST BE PERFORMED BY A QUALIFIED PILOT APPROVED FOR STALL TEST FLIGHTS BY LEARJET.**
- **ADJUST C.G. FOR STALL WARNING FLIGHT TEST FORWARD (FORWARD LIMIT TO 3% MAC AFT OF FORWARD LIMIT).**
- **FLY STALLS AT 8,000 TO 12,000 FEET (5,000 FEET AGL MINIMUM).**

NOTE:
- Use preceding "Stall Warning System Ground Adjustment" to make in-flight pusher shaker adjustments.
- Pusher and shaker points may require readjustment due to the interaction of the pusher and shaker.
- Stall warning light and stick shaker activation are simultaneous. At this point, shaker frequency is very low, requiring close attention to detect.
- Angle-of-attack indicator needles out of synchronization at high speeds reveal shaker or pusher actuation speeds are not coordinated between the pilot's and copilot's systems. Readjust 0° potentiometers to equalize the indications.

(1) Preflight

(a) Fabricate patch cables in accordance with Figure 202. The cables must be wired exactly as plug P85 on the computer-amplifier and plug P536 on the Stall Warning Bias Box. (Refer to applicable wiring diagram in the Wiring Manual.)

NOTE: There will be more wires in the patch cable bundle than there are pins in the frame No. 5 connector (3). Therefore, some of the grounds may be grouped by "jumpering" to reduce the quantity of wires to a number which the connector will accommodate.

(b) Remove the plug and plate from the patch cable access hole in aircraft frame 5 approximately LBL 4 and WL 22.

(c) Remove the Stall Warning Bias Box and the computer from the nose compartment and place in cabin area accessible for flight crew adjustments.

(d) Connect wire bundles in nose compartment to computer and bias box in cabin area with patch cables.

(e) Perform stall warning system functional test to prove integrity of patch cables. (Refer to Adjustment/Test, this section.)

(f) Perform nose steering system functional test to prove integrity of patch cables. (Refer to Chapter 32.)

(g) Fuel the aircraft wing and tip tanks.

(h) Weigh the aircraft. Add ballast as required so that the C.G. is at the forward limit (forward to 3% aft of forward) as shown on the stall speed charts and the Airplane Flight Manual C.G. chart.

(2) Flight Check at 8,000 to 12,000 Feet MSL (5,000 Feet AGL minimum)
  (a) In landing, takeoff, maneuvering, and cruise configuration, individually, with idle power, approach stall by decreasing airspeed at one (1) knot per second. Check each system separately.
  (b) Record fuel quantity and airspeeds at which shaker and pusher actuate.
  (c) Verify proper operation of stall warning "steady" and "flashing" lights.
  (d) Pusher actuation speeds shall fall on the stall speed curve +2, -0 knots for each configuration. Refer to FAA Approved Flight Manual (Section IV) for Stall Speeds. Enter Stall Speed chart using instrument corrected airspeed as provided by the correction card for the Pilot's Airspeed Indicator.
  (e) The shaker shall actuate at least 7% (in no case less than 7 knots) prior to the pusher actuation with gear and flaps fully down.
  (f) The angle-of-attack indicator needle should enter the red arc at pusher activation.
  (g) Adjust system as required until requirements of steps (d), (e), and (f) are met.
  (h) Recheck system. The pusher speed shall be within +2, -0 knots of the speeds reflected in the Flight Manual "Stall Speeds" curves.

(3) Postflight
  (a) Remove patch cables from aircraft. Install computer and bias box in nose compartment.
  (b) Install plug and plate in frame 5 access hole. Seal over pressure side of plate with Pro-Seal 890B.

CAUTION: DO NOT PERFORM ANY ADJUSTMENT OF COMPUTER PUSHER AND SHAKER ACTUATION POINTS AFTER COMPLETION OF THE FLIGHT ADJUSTMENT.

  (c) Repeat "Functional Test of Stall Warning System," paragraph 2.A., to verify that the existing vane plate markings for pusher and shaker match the flight-calibrated computer. If the vane plate markings match the computer, return aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to precisely match the computer pusher and shaker actuation points. (Refer to 27-31-10.)

D. Functional Test of Stall Warning System. (For Aircraft effectivity, refer to NOTE at beginning of Adjustment/Test.)

NOTE: Perform Functional Test of Stall Warning system in accordance with the current inspection interval specified in Chapter 5.

(1) With power off, check pilot's and copilot's angle-of-attack indicators needle position. The indicator needles shall be centered on the red/yellow intersection (±1/2 full needle width). If either needle is not centered, replace applicable indicator before performing remainder of test.

WARNING: AVOID CONTACT WITH HOT VANES.

(2) Set Battery Switches on, Pitot Heat Switches to L and R and verify that vanes are heated. Set Pitot Heat Switches to OFF.
(3) Momentarily set Left Stall Warning Switch to ON and ensure the left stall warning light extinguishes and the right stall warning light remains illuminated.
(4) Momentarily set Right Stall Warning Switch to ON and ensure the right stall warning light extinguishes and the left stall warning light remains illuminated.

(5) With angle-of-attack vanes completely lowered, set Stall Warning Switches to ON. Allow 15-minute warmup of stall warning computer. Verify that the following occurs:
   (a) L STALL and R STALL annunciators shall extinguish.
   (b) Master Warning annunciators shall extinguish.
   (c) Pilot's and copilot's angle-of-attack indicator needle shall move in to the green area.
   (d) Neither shaker nor pusher shall actuate, even transiently (a slight tremor at the control wheel is acceptable).

(6) Set Stall Warning Switches to OFF. L STALL and R STALL annunciators shall illuminate and Master Warning annunciators shall flash.

(7) Actuate both squat switches to the air mode (aircraft on jacks) and set Stall Warning Switches to L and R. Verify flaps at 0°.

(8) Set TEST Switch to L STALL and depress pushbutton. No response shall be noted.

(9) Set TEST Switch to R STALL and depress pushbutton. No response shall be noted.

(10) Actuate squat switches to ground mode. Set TEST switch to L STALL. Verify that the following occurs:
   (a) Pilot's angle-of-attack indicator needle shall start moving from green area toward the red area at a slow rate.
   (b) As the indicator needle moves into the yellow area, the L STALL annunciator shall flash, Master Warning annunciators shall flash, and the LH stick shaker shall actuate (there may be a slight lag as the shaker motor comes up to speed).
   (c) As the indicator needle reaches the red line, the L STALL annunciator flashing shall become steady momentarily and pusher shall actuate briefly then automatically stop as the indicator needle falls back into the yellow or green area. Release pushbutton and verify that indicator needle falls back into the green area.

(11) Repeat steps (10)(a) through (c) for the RH stall warning system with the TEST Switch set to R STALL.

(12) Set TEST Switch off.

NOTE: On *Aircraft 35-506 and Subsequent and 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot,"* control column nudger will actuate simultaneously with the shaker. The nudger is evidenced by a pulsing of the control column (1 1/2 - 2 1/2 times per second) forward (nose down) by the pitch servo.

(13) Extend flaps full down.

NOTE:
- The angle-of-attack vane to vane plate mark tolerance shall be +0, -1/32 inch (+ 0, - 0.79 millimeter) for pusher operation and ± 1/16 inch (± 1.59 millimeter) for shaker operation. A positive (+) error means the vane is above the appropriate vane plate mark. A negative (-) error means the vane is below the appropriate vane plate mark.

- When performing step D.(15) the actuation for pusher at 40° and 20° of flap setting, for some aircraft, occur at the same point. For these aircraft the upper line (pusher) is identified by ••• and is used to check for pusher actuation at 40° and 20° flap settings.

(14) Slowly lift left angle of attack vane until the aft lower edge of vane is centered on the lower line marked ••••. The shaker shall actuate, the L STALL warning light will flash, and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.

(15) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ••••. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.

(16) Retract flaps to 20°.
(17) Slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked •••. The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(18) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked •••. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(19) Retract flaps to 8°.
(20) Slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked ••. The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(21) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ••. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(22) Retract flaps to full up (verify that flaps are all the way up).
(23) Slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked •. The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(24) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked •. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(25) Repeat steps (13) thru (24) using the right stall warning switch, right angle-of-attack vane, copilot's angle-of-attack indicator, and R STALL warning light.
(26) If the stall warning system complies with this functional test, perform step (28) and return the aircraft to service.
(27) If the stall warning system fails to comply with this functional test, proceed as follows:
  (a) Check flap position switch rigging. (Refer to 27-50-05 or 27-51-03.)
  (b) On *Aircraft 35-067 thru 35-278 and 36-018 thru 36-044 not equipped with stall fence (AAK 79-10 or AMK 83-5), nor modified per AAK 83-2, "Installation of FC-530 Autopilot," the "Stall Warning System Ground Adjustment"* per paragraph 2.E. and the "Stall Warning System Flight Adjustment/Test" per paragraph 2.F. must be performed.
  (c) On *Aircraft 35-279 and Subsequent and 36-045 and Subsequent and prior Aircraft equipped with stall fence (AAK 79-10 or AMK 83-5 or modified per AAK 83-2, "Installation of FC-530 Autopilot,"*) check vane plate marking. (Refer to 27-31-10.) Then elect to either (1) perform the "Stall Warning System Ground Adjustment" per paragraph 2.E. and the "Stall Warning System Flight Adjustment/Test" per paragraph 2.G. or (2) perform the "Stall Warning System Ground Calibration" per paragraph 2.H.
(28) Set Battery Switches and Stall Warning Switches to OFF. Install end cover on computer. Install nose compartment doors. Remove aircraft from jacks unless jacks required for subsequent procedure.

E. Stall Warning System Ground Adjustment (See Figure 203.) (For Aircraft effectivity, refer to NOTE at beginning of Adjustment/Test.)

NOTE:
- The pusher-shaker systems should be adjusted only by completely qualified personnel, preferably at a factory-authorized service center.
- Any adjustment of the pusher will require readjustment of the corresponding shaker point.
- Angle-of-attack vanes must be moved very slowly to give correct indication because vane rate is used in stall computation.

(1) Remove nose compartment access doors.
(2) Remove end cover from computer to gain access to adjustment potentiometers.
(3) Connect auxiliary power unit to aircraft to supply 28 vdc for test adjustment procedures.
(4) Retract flaps to full up.
(5) Set Stall Warning Switches to OFF.
(6) Set Battery Switches on.
(7) Ensure that both angle-of-attack vanes are full down.
(8) Set left Stall Warning Switch to L. Allow 15-minute warm-up of stall warning computer.
(9) Slowly lift left angle-of-attack vane until pilot's angle-of-attack indicator needle is centered at the intersection of the green and yellow bands. The shaker shall actuate and L STALL warning light will begin to flash. If necessary, adjust left RX potentiometer until shaker and warning light actuation occurs at this point.
(10) Slowly lower angle-of-attack vane. The shaker will stop and warning light will extinguish.
(11) Slowly raise angle-of-attack vane. Ensure that shaker and warning light energize when angle-of-attack indicator needle is at the intersection of the green and yellow bands.
(12) Extend flaps full down.
(13) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ••••. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line. If necessary, adjust the left pusher 40° potentiometer until pusher actuates at this point.
(14) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line marked ••••. The shaker shall actuate, the L STALL warning light will flash. The pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands. If necessary, adjust the left shaker 40° potentiometer until shaker and warning light actuate at this point.
(15) Retract flaps to 20°.
(16) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked •••. The pusher should actuate. The pilot's angle-of-attack indicator should be at the red line. If necessary, adjust the left pusher 20° potentiometer until pusher actuates at this point.
(17) Slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked •••. The shaker and L STALL warning light shall actuate. The pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands. If necessary, adjust the left shaker 20° potentiometer until shaker and warning light actuate at this point.
(18) Retract flaps to 8°.

(19) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ••. The pusher shall actuate. The pilot's angle-of-attack indicator shall be at the red line. If necessary, adjust the left pusher 8° potentiometer until pusher actuates at this point.

(20) Slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked ••. The shaker and L STALL warning light shall actuate. The pilot's angle-of-attack indicator needle shall be at the intersection of the green and yellow bands. If necessary, adjust the left shaker 8° potentiometer until shaker and warning light actuate at this point.

(21) Retract flaps to full up (verify that flaps are all the way up).

(22) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked •. The pusher shall actuate. The pilot's angle-of-attack indicator needle shall be at the red line. If necessary, adjust the left pusher 0° potentiometer until pusher actuates at this point.

(23) Slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked •. The shaker and L STALL warning light shall actuate. The pilot's angle-of-attack indicator needle shall be at the intersection of the green and yellow bands. If necessary, adjust the left shaker 0° potentiometer until shaker and warning light actuate at this point.

(24) Repeat steps (8) thru (23) using the right stall warning switch, right angle-of-attack vane, copilot's angle-of-attack indicator, right stall warning potentiometers, and R STALL warning light.

F. Stall Warning System Flight Test/Adjustment (For Aircraft effectivity, refer to NOTE at beginning of Adjustment/Test.)

WARNING:

- **STALL WARNING SYSTEM FLIGHT TESTING MUST BE PERFORMED BY A QUALIFIED PILOT APPROVED FOR STALL TEST FLIGHTS BY LEARJET INC.**
- **ADJUST C.G. FOR STALL WARNING FLIGHT TEST FORWARD (FORWARD LIMIT TO 3% MAC AFT OF FORWARD LIMIT).**
- **FLY STALLS AT 8,000 TO 12,000 FEET (5,000 FOOT AGL MINIMUM).**

NOTE:

- Use preceding "Stall Warning System Ground Adjustment" as a guide to make in-flight pusher shaker adjustments. The "Stall Warning System Ground Adjustment" shall be completed prior to the flight calibration.
- Make pusher adjustment prior to shaker adjustment. Adjustment of pusher circuitry will affect shaker setting.
- Stall warning light and stick shaker activation are simultaneous. At this point shaker frequency is very low, requiring close attention to detect.
- Angle-of-attack indicator needles out of synchronization at high speeds reveal shaker or pusher actuation speeds are not coordinated between the pilot's and copilot's systems. Readjust 0° potentiometers to equalize the indications.

(1) Visually inspect the following systems and components for security of mounting and general condition.
   (a) Flaps, spoilers, and ailerons.
   (b) Wheel well doors.
   (c) Tip tank strakes.
   (d) Wing leading edge (inspect particularly for dents).
   (e) Pitot static ports for condition.

(2) Check flap system for proper rigging. Also check that flap position indicator is properly calibrated. (Refer to 27-50-01.)

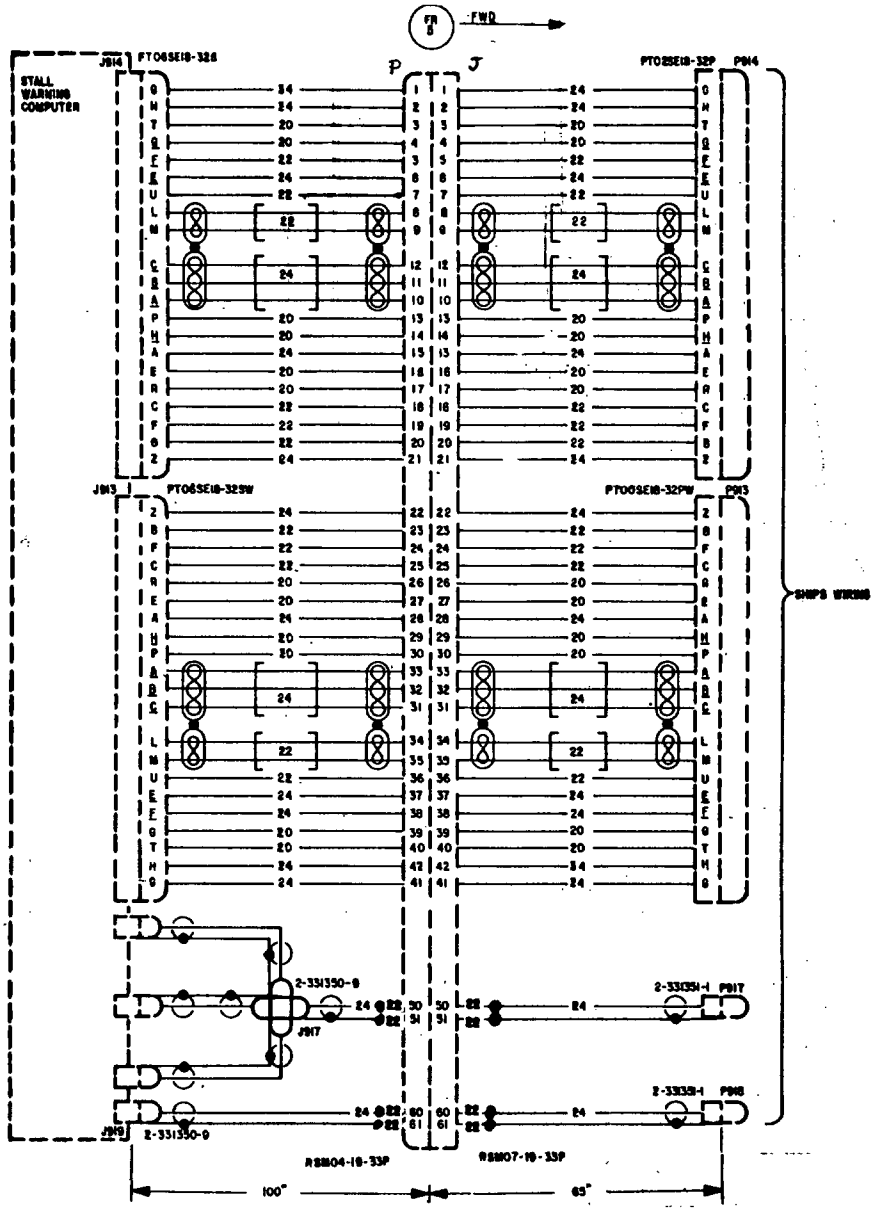
NOTE: Exert sufficient upward load on the flaps by hand to remove any possible free play when checking flap positions.

(3) Check the spoiler and aileron system rigging.
(4) Check the pilot and copilot airspeed indicators as follows:
(a) Perform Pitot System Leakage check using a pitot static test set. Ensure that leakage rate is not exceeded. Connect test set to both pitot tubes so that both the pilot and copilot airspeed indicators can be checked.

NOTE: Start with a higher pitot pressure than each target pressure. Reduce the pressure until each target pressure is reached.

(b) Adjust the pitot static tester so that tester airspeed indicates above 108 knots.
(c) Adjust tester to target airspeed 105 knots. Record pilot and copilot airspeed readings.
(d) Adjust tester to 95 knots and 85 knots and record each pilot and copilot airspeed reading.
(e) Refer to tester airspeed indicator and pilot and copilot airspeed indicator correction cards. Pilot and copilot correct airspeed indication shall be within ± 2 knots of corrected airspeed indication at each target setting.
(5) Perform stall warning preflight weight and balance adjustments as follows:
(a) Fill the wing and tip fuel tanks.
(b) Weigh the aircraft. Add ballast as required so that the C.G. is at the forward limit (forward limit to 3% MAC aft of forward limit) as shown on the stall speed charts and the Airplane Flight Manual C.G. chart.
(6) Ensure that Stall Warning Switches are set to OFF.
(7) Remove plug and plate from patch cable access hole. The plug is located on frame 5 at RBL 4, WL 22 or 5 inches (12.7 centimeters) below outflow valve.
(8) Remove stall warning computer from nose compartment and place it in the cabin.
(9) Install patch cable connectors into frame 5 bulkhead and connect to ship's wiring and computer. Patch cable P/N 15-1013 is available from Learjet Spares Department.
(10) Secure nose compartment access doors.
(11) Remove computer end cover.
(12) Perform stall warning functional test to prove integrity of patch cable.
(13) Record aircraft takeoff weight. Approach stall speed by decreasing airspeed at one (1) knot per second. Check each system separately. Fly the aircraft in each of the following configurations:
Gear Down - Flaps Down (40°)
Gear Up/Dn - Flaps 20°
Gear Up/Dn - Flaps 8°
Gear Up/Dn - Flaps Up (0°)
(a) Pusher actuation speed shall fall on stall speed curve +2;-0 knots for each configuration. Refer to Approved Flight Manual performance data for stall speeds. Enter stall speed charts using instrument-corrected airspeed as provided by correction card for Pilot's Airspeed Indicator. Pusher must actuate before aerodynamic stall.

LIST OF MATERIALS		
ITEM NO. (or equivalent)	QTY	DESCRIPTION
2-331350-9	2	P917 & P918
PTO6SE18-32SW	1	P913
PTO6SE18-32S	1	P914
RSM04-19-33S	1	Plug-Bulkhead
2-331351-1	2	J917 & J918
PTOOSE18-32PW	1	J913
PTO2SE18-32P	1	J914
RSM07-19-33P	1	Recept.-Bulkhead
MIL-W-81044/2 MIL-W-16878 Type B	165'	20 AWG Wire
	124'	22 AWG Wire
BUB1936NAL63 or MB1044/4-22-9	152'	24 AWG Wire
1SBUB1936NAL-63-JN MIL-W-16878 Type E	28'	24 AWG Wire Two 22 AWG Teflon Wires twisted and shielded with fiberglass braid
3SBUB1936NAL-63-JN	28'	Three 22 AWG Wires twisted and shielded with Nylon Jacket



Fabrication and Use of Patch Cables
Figure 204

(b) Shaker shall actuate at least 7% (in no case less than 7 knots) above the pusher speed in all configurations.
(c) Angle-of-attack needle shall enter the yellow band at shaker and the red band at pusher.
(d) Adjust system as required until requirements of steps (13)(a), (b), and (c) are met.

NOTE: Roll characteristics shall remain acceptable to pusher speed (no roll-off beyond 20° bank).

(e) Recheck system. Pusher speed shall be within +2;-0 knots of speeds reflected in the Approved Airplane Flight Manual stall speed charts.

(14) After the pusher-shaker has been adjusted, approach a stall at a rate greater than 1 knot per second. The angle-of-attack rate feature shall actuate the pusher before aerodynamic stall.
(15) Climb to 22,500 (±750) feet.
(a) Pilot and copilot angle-of-attack indicator needles shall abruptly move toward yellow band, indicating correct altitude switch function.
(b) Match indication at cruise by adjusting the higher indicator reading to match the lower. Recheck stall pusher and shaker at zero degrees. (Refer to step (13) above.)
(16) Postflight, remove patch cable and install stall warning computer in nose avionics compartment. ensure that plugs are bottomed in receptacles. Secure with attaching parts.
(17) Replace frame 5 access hole plug and plate. Seal pressure side with ProSeal 890B.

CAUTION: DO NOT PERFORM ANY ADJUSTMENTS OF COMPUTER PUSHER AND SHAKER ACTUATION POINTS AFTER COMPLETION OF THE FLIGHT CALIBRATION.

(18) Repeat "Functional Test of Stall Warning System," paragraph 2.D. (steps 12 thru 25), to verify that the existing vane plate markings for pusher and shaker match the flight calibrated computer. If the vane plate markings match the computer, return aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to precisely match the computer pusher and shaker actuation points before returning the aircraft to service. (Refer to 27-31-10.)

G. Stall Warning System Flight Test/Adjustment (For Aircraft effectivity, refer to NOTE at beginning of Adjustment/Test.)

WARNING: STALL WARNING SYSTEM FLIGHT TESTING MUST BE PERFORMED BY A QUALIFIED PILOT APPROVED FOR STALL TEST FLIGHTS BY LEARJET CORPORATION.

NOTE:
- Use preceding "Stall Warning System Ground Adjustment," paragraph 2.E. as a guide to make inflight pusher shaker adjustments. The "Stall Warning System Ground Adjustment" shall be completed prior to the flight calibration.
- Make pusher adjustments prior to shaker adjustment. Adjustment of pusher circuit will affect shaker setting.
- Stall warning light and stick shaker activation are simultaneous. At this point, shaker frequency is very low, requiring close attention to detect.
- Angle-of-attack indicator needles which are out of synchronization at high speeds reveal that shaker or pusher actuation speeds are not coordinated between the pilot's and copilot's systems. Readjust CLEAN potentiometers to equalize the indications.

(1) Visually inspect the following systems and components for security of mounting and general condition:
   (a) Flaps, spoilers, and ailerons.
   (b) Wheel well doors.
   (c) Tip tank strakes.
   (d) Wing leading edge (inspect particularly for dents).
   (e) Pitot static ports for condition.

(2) Check flap system for proper rigging. Also check flap position indicator for proper indication. (Refer to 27-50-02.)

   NOTE: Exert sufficient upward load on the flaps by hand to remove any possible free play when checking flap positions.

(3) Check the spoiler and aileron system rigging.

(4) Check the pilot and copilot airspeed indicators as follows:
   (a) Perform Pitot System Leakage Check using a pitot-static test set. Ensure that leakage rate is not exceeded. Connect test set to both pitot tubes so that both the pilot and copilot airspeed indicators can be checked.

      NOTE: Start with a higher pitot pressure than each target pressure. Reduce pressure until each target pressure is reached.

   (b) Adjust pitot-static tester so that tester airspeed indicator indicates above 108 knots.
   (c) Adjust the tester to the target airspeed 105 knots. Record pilot and copilot airspeed ratings.
   (d) Adjust tester to 95 knots and 85 knots and record each pilot and copilot airspeed reading.
   (e) Refer to tester airspeed indicator and pilot and copilot airspeed indicator correction cards. Pilot and copilot corrected airspeed indication shall be within ± 2 knots of corrected airspeed indication, at each target setting.

(5) Perform stall warning preflight weight and balance adjustments as follows:
   (a) Fill the wing and tip fuel tanks.
   (b) Weigh the aircraft. Add ballast as required so that the C.G. is at the forward limit (forward to 1.54% aft of forward) as reflected in the Approved Airplane Flight C.G. chart and stall speed charts.

(6) Ensure that Stall Warning Switches are set to OFF.

(7) Remove plug and plate from patch cable access hole on frame 5 at RBL 4, WL 22, or 5 inches (12.7 centimeters) below outflow valve.

(8) Remove stall warning computer from nose compartment and place it in the cabin.

(9) Install patch cable connectors into frame 5 bulkhead and connect to ship's wiring and computer. Patch cable P/N 15-1013 is available from Learjet Spares Department.

(10) Secure nose compartment access doors.
(11) Remove computer end cover.
(12) Perform "Functional Test of Stall Warning System," paragraph 2.D. (steps 12 thru 25), to prove integrity of patch cable. (See Figure 204.) Allow 15-minute warm-up time for stall warning computer.
(13) Flight test aircraft as follows:
  (a) The stall warning computer will be calibrated with the aircraft configured as follows:
    Gear Up - Flaps Up (0°)
    Gear Down - Flaps 8°
    Gear Down - Flaps 20°
    Gear Down - Flaps Down (40°)

    NOTE: Flight test aircraft at 8,000 to 15,000 feet (5,000 foot AGL minimum).

  (b) Record aircraft takeoff weight.
  (c) Check angle-of-attack flap biasing at 8°, 20°, and 40° as follows:
    1) Select 8° of flap. As flaps pass 3°, indicator needle shall move toward red band.
    2) Select 20° of flap. As flaps pass 13°, a switching transient shall be noted, but final positions shall be same as 8° setting.
    3) Select 40° of flap. As flaps pass 25°, a switching transient shall be noted, but final positions shall be same as 8° setting.
  (d) Determine Chart Stall Speed for each configuration. Trim to 1.3 times stall speed.

    WARNING: DO NOT STALL THE AIRCRAFT AT ANY C.G. MORE THAN 1.54% AFT OF FORWARD WITH STALL WARNING SWITCHES SET TO OFF (PUSHER OFF).

  (e) Set Stall Warning Switches to OFF. At approximately 12,000 feet, gradually increase back pressure and fly the aircraft to aerodynamic stall at a deceleration rate of 1 knot per second. Note corrected stall speed. Aircraft will remain controllable in roll until aerodynamic stall/aft stick is reached. No roll-off of greater than 20° bank is allowable throughout stall speed recovery.
  (f) Record aerodynamic stall/aft stop speed for the configuration being calibrated. Actual aerodynamic stall speed must not exceed by more than 3 knots the stall speed reflected in the Approved Airplane Flight Manual stall speed charts.
  (g) With Stall Warning Switches ON, set stall warning system pusher to actuate at ±3 knots of published stall/pusher speed in each configuration. Pusher must actuate at or above aerodynamic stall.
  (h) Set shaker to actuate a minimum of 7 knots or 7% above pusher speed.
  (i) After setting pusher/shaker for each configuration, record pusher/shaker speed and gross weight at that configuration.

    NOTE: Roll characteristics shall remain acceptable to pusher speed.

(14) After pusher/shaker has been adjusted, approach the stall at a rate greater than 1 knot per second. The angle-of-attack rate shall actuate the pusher prior to aerodynamic stall.
(15) Climb to 22,500 (±750) feet.
  (a) Pilot and copilot angle-of-attack indicator needles shall abruptly move toward yellow band, indicating correct altitude switch function.
(16) Postflight, remove patch cable and install stall warning computer in nose avionics compartment. Ensure that plugs are bottomed in receptacles. Secure with attaching parts.
(17) Replace frame 5 access hole plug and plate. Seal pressure side with Pro-Seal 890B.

CAUTION: DO NOT PERFORM ANY ADJUSTMENTS OF COMPUTER PUSHER AND SHAKER ACTUATION POINTS AFTER COMPLETION OF THE FLIGHT CALIBRATION.

(18) Repeat "Functional Test of Stall Warning System" paragraph 2.D. (steps 12 thru 25), to verify that the existing vane plate markings for pusher and shaker match the flight calibrated computer. If the vane plate markings match the computer, return aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to match the computer pusher and shaker actuation points prior to returning aircraft to service. (Refer to 27-31-10.)

H. Stall Warning System Ground Calibration (*Aircraft 35-279 and Subsequent, 36-045 and Subsequent, and prior aircraft equipped with stall fence [AAK 79-10 or AMK 83-5] or modified per AAK 83-2, "Installation of FC-530 Autopilot".*) (See Figure 205.)

NOTE:
- This calibration procedure shall be performed when a transducer, computer, or indicator is replaced or if the stall warning system has failed the Functional Test of Stall Warning System (paragraph 2.D.). Replacing the angle-of-attack vane requires performing the test per paragraph 2.D. (steps 12 thru 25).
- When replacing the indicator, if a quick check shows the indicator to be at yellow/green intersection for shaker and at red/yellow for pusher, further checks are not required.
- The Stall Warning System Ground Adjustment and Flight Adjustment/Test procedures (paragraphs 2.E. and 2.G.) can be used in lieu of these procedures.

(1) Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Inclinometer	T3500002-600 (Tool No. 950016)	Learjet Inc. Wichita, KS	Calibration procedure.
Stall Warning Ground Calibration Test Set	5471100-7 (Tool No. TS1100D)	Learjet Inc. Wichita, KS	Calibration procedure.
Mylar Template	5471102-7 (LH) 5471102-8 (RH)	Learjet Inc. Wichita, KS	Calibration procedure.
Digital Voltmeter		Commercially Available	Measure voltage.

(2) Set BAT 1 and BAT 2 Switches and L STALL and R STALL Switches to OFF.
(3) Place aircraft on jacks. Ensure that wheels are clear of ground.
(4) Level the aircraft in the pitch axis (refer to Chapter 8) or place inclinometer tool on cabin seat rail with FWD arrow pointing forward. Ensuring that inclinometer is free of obstructions, raise or lower nose jack to center inclinometer bubble.

NOTE: Move inclinometer to other areas of seat rails to verify leveling. A long precision bar (18 to 36 inches [25.4 to 91 centimeters]) can be used to average uneven area of the seat rails.

(5) Place a piece of tape forward of LH angle-of-attack vane so that rubber-tipped tension shaft of angle-of-attack vane inclinometer will not mar painted surface. Install vane inclinometer on LH angle-of-attack vane. Secure tool finger-tight only and release tension shaft to the extended position.

NOTE: Locate inclinometer on angle-of-attack vane so that approximately 0.25 inch (6.3 millimeters) gap exists between the retracted tension shaft rubber tip and aircraft skin.

(6) Remove nose compartment access doors.
(7) Disconnect electrical connectors and coax connectors from stall warning computer.
(8) Connect patch cable (W2, LH side) between test box , stall warning computer (LH side), and aircraft wiring (P913, LH side).

NOTE: No connections are made to coaxial cables.

(9) Connect Digital Voltmeter (DVM) to the test terminals (J2 and J3) on back of Stall Computer test box.
(10) Remove attaching parts and cover (end opposite electrical connectors) from stall warning computer.
(11) Set BAT 1 and BAT 2 and L STALL WARNING Switches on. Check that PWR annunciator on test box is illuminated and that SQ SW annunciator is extinguished. Block LH squat switches to the GND mode. Check that SQ SW annunciator is illuminated. Allow 30-minute warm-up of stall warning computer.
(12) Set test box RX CAL Switch to 35 position.
(13) Adjust CAL VANE LEVEL as follows:
  (a) Rotate Calibration Flap Selector Knob to CAL.
  (b) Set CAL Selector Switch to VANE LEVEL.
  (c) Level LH angle-of-attack vane at 0° using the inclinometer. Release tension shaft to hold angle-of-attack vane in this position.
  (d) Adjust CAL VANE LEVEL potentiometer to obtain zero (null) reading on null meter.
  (e) Set and hold SENS Switch to NULL. Adjust CAL VANE LEVEL potentiometer to obtain zero (null) reading on null meter. Release SENS Switch.
(14) Adjust CAL VANE HI as follows:
  (a) Set CAL Selector Switch to VANE HI.
  (b) Pull out on inclinometer tension shaft and position LH angle-of-attack vane at 33-1/3 degrees. Release tension shaft to hold angle-of-attack vane in this position.
  (c) Adjust CAL VANE HI potentiometer to obtain zero (null) reading on null meter.
  (d) Set and hold SENS Switch to NULL. Adjust CAL VANE HI potentiometer to obtain zero (null) reading on null meter. Release SENS Switch.
(15) Repeat steps (13) and (14) until no further adjustment is required. Null meter shall be 0 (±4) marks with SENS Switch set to NULL.

NOTE: Do not disturb these calibration adjustments during the remainder of the test for the LH system.

Adjust potentiometers very slowly to ensure accurate results.

Aircraft static system must be lower than 22,500 feet during the following adjustments.

(16) Rotate Calibration Flap Selector Knob to RX.
(17) Rotate digital potentiometer for an indication of 100 full right on the test set meter.
(18) Verify that DVM reads 1.95 (±0.05) vdc. If not, slowly adjust test set digital potentiometer to achieve 1.95 vdc reading on the DVM.

(19) Adjust left stall system RX potentiometer until test box SHAKE and WARN annunciators are just at the threshold of blinking.

NOTE: Do not disturb adjustment of RX potentiometer after completion of step (19).

(20) Rotate Calibration Flap Selector Knob to 0°. Rotate digital potentiometer to 0° pusher value. (Refer to Table 1, Figure 205.)

NOTE: Adjust pusher potentiometer first, then shaker. If any pusher potentiometer is readjusted, then the corresponding shaker potentiometer must also be readjusted.

(21) Adjust left stall system 0° PUSHER potentiometer for threshold illumination of test box PUSH annunciator. Null meter shall read approximately zero.
(22) Rotate digital potentiometer below 0° pusher value. Threshold illumination must occur within ±2 units of the clean pusher value as the digital potentiometer setting is increased. Repeat steps (20) and (21) if necessary.
(23) Rotate digital potentiometer to 0° shaker value. (Refer to Table 1, Figure 205.)
(24) Adjust left stall system 0° SHAKER potentiometer for threshold blinking of the test box SHAKE and WARN annunciators. Null meter shall read approximately 40 right.
(25) Rotate digital potentiometer below the 0° shaker value. Threshold blinking shall occur within ±2 units of 0° shaker value.
(26) Repeat steps (20) thru (25) using the Calibration Flap Selector Knob at the remaining flap settings (8°, 20°, and 40°).
(27) Using the values in Table 1, Figure 205, set digital potentiometer to each value at each flap setting and verify SHAKE, WARN, and PUSH annunciators blink within ±2 units. If necessary, adjust potentiometers.

NOTE: Changing any pusher adjustment requires that the corresponding shaker potentiometer also be adjusted.

(28) Set Calibration Flap Selector Knob to 0°.
(29) Set digital potentiometer to cruise setting (Table 1, Figure 205) and verify that pilot's angle-of-attack indicator (P/N 6608208) needle is at 3rd mark (± 1/4 needle width) from green and yellow intersection in green area. On *Aircraft equipped with angle-of-attack indicator (P/N 6600082-6)*, the indicator needle shall be at 4.5 marks (± one needle width) from green and yellow intersection in green area. Adjust left stall system 0° SHAKER potentiometer if necessary.

NOTE: Final adjustment must result in a 0° shaker activation within ± 15 units of the 0° shaker value specified in Table 1, Figure 205.

(30) Set digital potentiometer to a test set meter indication of 50 on right side of zero. Depress and hold ALT Switch and verify null meter deflection to the left.
(31) Cycle aircraft flaps to 0°, 8°, 20°, and 40° positions. As the flaps pass through 3°, 13°, and 25°, the respective FLAPS annunciators on the test box shall illuminate.
(32) Remove vane tool from LH angle-of-attack vane and install on RH angle-of-attack vane.
(33) Set Battery Switches and Stall Warning Switches to OFF.
(34) Disconnect patch cable (W2, LH side) from test box, computer and aircraft wiring.
(35) Connect patch cable (W1, RH side) between test box, computer (RH side) and aircraft wiring (P914, RH side). Repeat steps H.(14) thru H.(33) for the RH stall warning system.
(36) Set Battery Switches and Stall Warning Switches off. Position cover on end of stall warning computer and secure with attaching parts.
(37) Disconnect patch cable (W1, RH side) from test box, computer, and aircraft wiring. Connect electrical connectors and coax cables to computer.

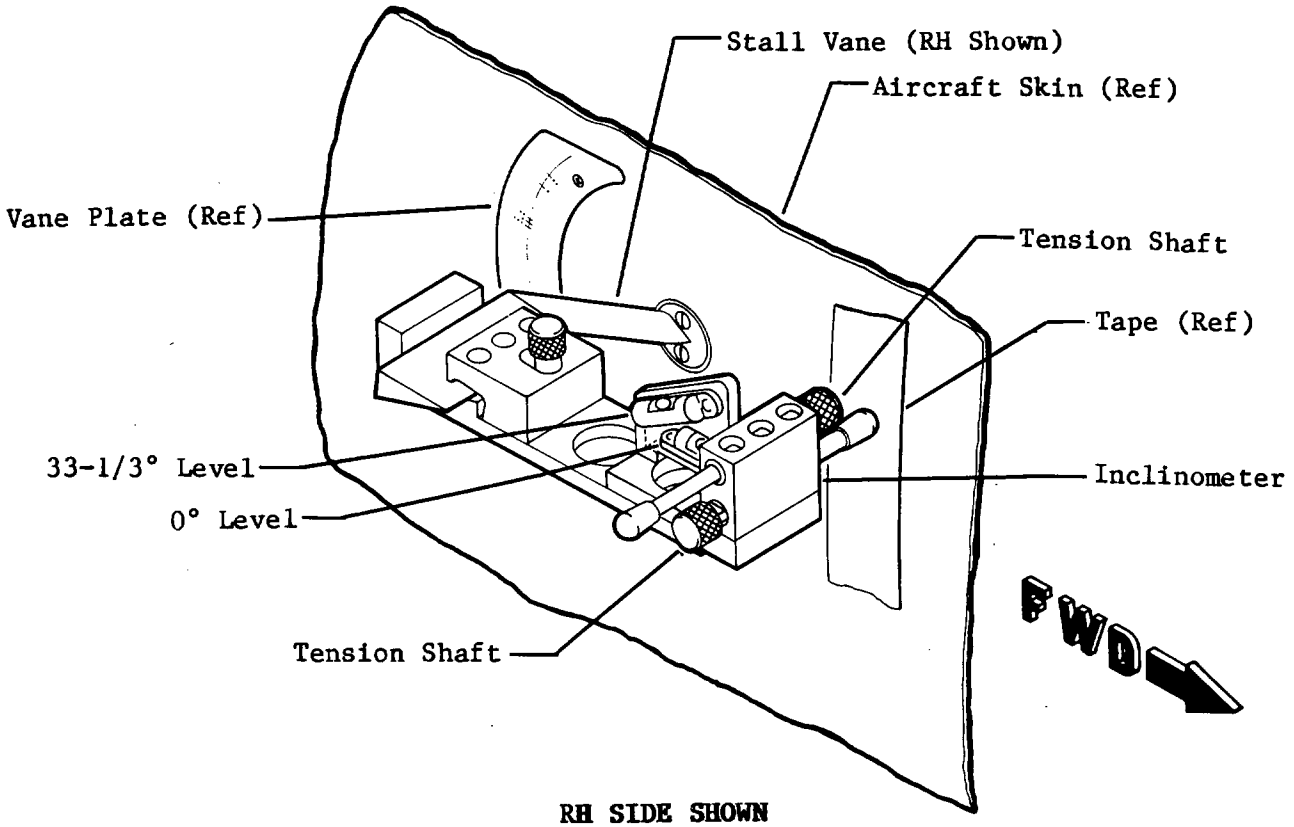
Calibration Flap Selector Switch Setting	Digital Pot Setting*	
	Pusher ±2	Shaker ±2
0°	600	435
8°	735	495
20°	705	480
40°	720	450
Cruise Setting	094	

* Effective Aircraft 35-279 thru 35-505, 36-045 thru 36-053 and prior aircraft modified per AAK79-10 or AMK83-5, "Stall Fence", but not modified per AAK83-2, "Installation of FC-530 Autopilot".

Calibration Flap Selector Switch Setting	Digital Pot Setting**	
	Pusher ±2	Shaker ±2
0°	600	435
8°	855	540
20°	855	540
40°	855	510
Cruise Setting	094	

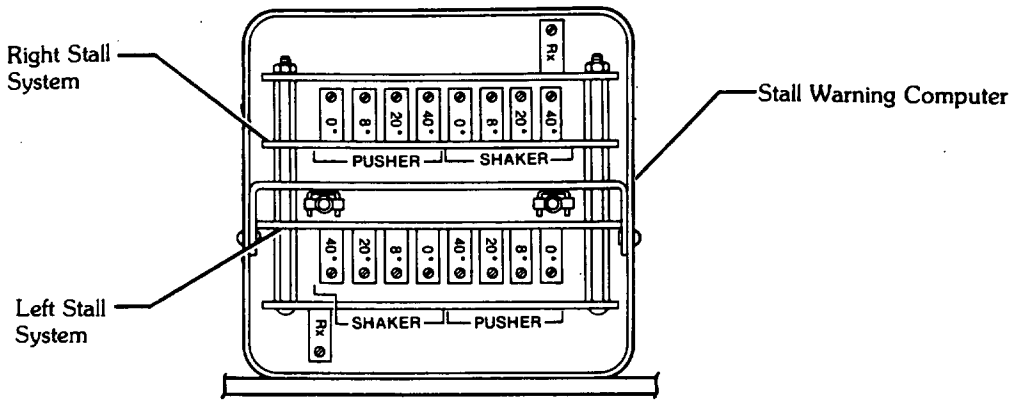
** Effective Aircraft 35-408, 35-506 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per AAK83-2, "Installation of FC-530 Autopilot".

TABLE 1

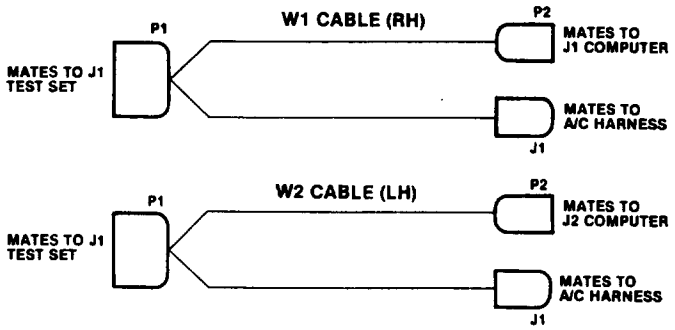


14-166A

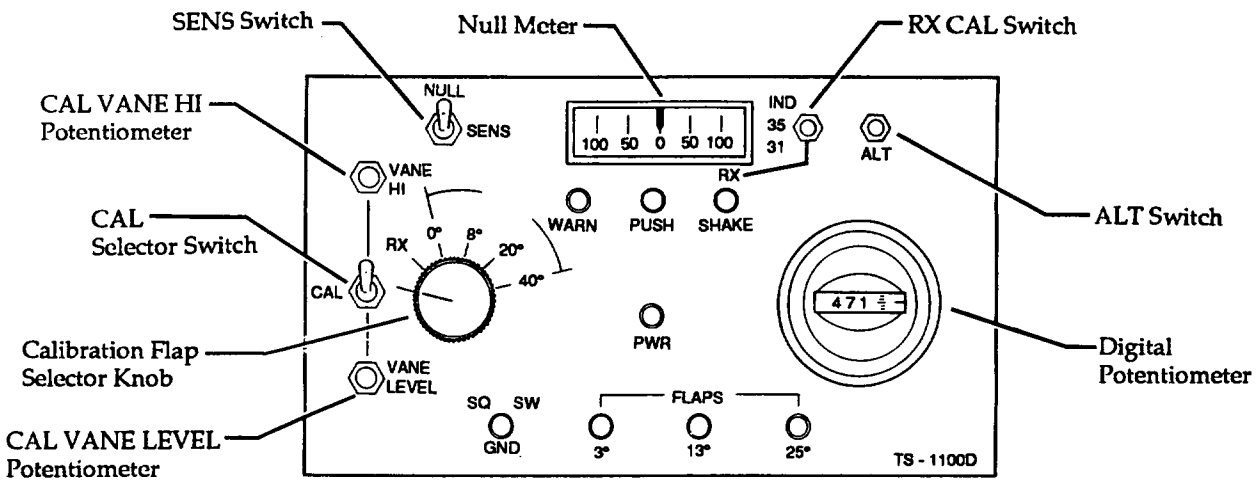
Stall Warning System Ground Calibration
Figure 205 (Sheet 1 of 2)



END OPPOSITE PLUG SHOWN
COVER REMOVED



Patch Cable



14-172B
9-343A

Stall Warning System Ground Calibration
Figure 205 (Sheet 2 of 2)

(38) Perform Functional Test of Stall System, paragraph 2.D. (steps 12 thru 28), to verify that the existing vane plate markings for pusher and shaker match the calibrated computer. If the vane plate markings match the computer, return aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to match the computer pusher and shaker actuation points prior to returning aircraft to service. Use the mylar templates as an aid to ensure that marks and calibration are correct. (Refer to 27-31-10.)

I. Calibration Test of Stall Warning Ground Calibration Test Box

(1) Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Digital Multimeter	8000 Series	Fluke Mfg. Inc. Everett, WA	Measure voltage and resistance.
0-28 VDC Variable Power Supply		Commercially Available	Provide elect. power for test.

CAUTION: PERFORM ALL STEPS OF THE CALIBRATION TEST IN A SEQUENTIAL MANNER. FAILURE TO DO SO MAY RESULT IN DAMAGE TO UNIT UNDER TEST.

NOTE: This calibration test is to be performed annually to ensure the integrity of the ground calibration test box (P/N 5471100-7). If the ground calibration test box fails the following test, it must be returned to Learjet Inc. for recalibration.

(2) Confirm continuity checks by inserting the (+) positive and (-) negative leads of the ohmmeter into the indicated pins of the test set connector as shown in Table 2. Verify continuity is less than one (1) ohm.

(+) Lead	(-) Lead
Pin L	Pin p
Pin M	Pin q
Pin a	Pin g
Pin D	Pin P
Pin c	Pin s
Pin n	Pin H
Pin G	Pin Y
Pin w	Pin X
Pin X	Pin m
Pin m	Pin y
Pin y	Box Gnd
+ Test Point	Pin Y
- Test Point	Pin H
Pin E	Pin k

Table 2

(3) Insert (+) positive lead of ohmmeter into pin L and (-) negative lead into pin M. There shall be no continuity. Depress and hold ALT Switch. Verify that continuity exists. Release ALT Switch.
(4) Rotate Calibration Flap Selector Switch to RX and set RX CAL Switch to IND. Verify continuity between pins Y and H.
  (a) Rotate Calibration Flap Selector Switch from 0° thru 40° positions. Verify ohmmeter reading of 30 (±1.5) K ohms at each position.
(5) Rotate Calibration Flap Selector Switch to CAL, set CAL Selector Switch to VANE LEVEL and rotate digital potentiometer to full CCW. Verify ohmmeter reading of 30.1 (±1.6) K ohms between pin B and R.
(6) Rotate Calibration Flap Selector Switch to RX position and set RX CAL Switch to the 35 position. Verify open circuit between pin B and pin Y.
  (a) Set and hold SENS Switch to NULL. Verify ohmmeter reading of 15 K to 23 K ohms.
(7) Adjust VANE HI potentiometer full CW and the VANE LEVEL potentiometer full CCW. Verify ohmmeter reading of 1360 (±140) ohms between pins A and D.
(8) Using the Calibration Flap Selector Switch verify the following conditions:
  (a) Switch at 8 ° FLAP position; check for continuity between pins D and U.
  (b) Switch at 20° FLAP position; check for continuity between pins D, U and U.
  (c) Switch at 40° FLAP position; check for continuity between pins D, U, U and V.
(9) Remove any ohmmeter connections. Rotate Calibration Flap Selector Switch to RX position.
(10) Set variable power supply to 1.95 (±0.05) vdc.

CAUTION: DO NOT SET RX CAL SWITCH TO IND POSITION WHILE PERFORMING STEPS 11 THRU 16. PERFORM STEPS IN A SEQUENTIAL MANNER.

(11) Connect variable power supply (+) positive to pin Y and (-) negative to pin H.
(12) Verify that Null Meter is at the full scale indices ±1/2 needle width.

NOTE: View indicator from a minimum of two feet with eye at center of meter. Power supply voltage may be verified using (+) and (-) test points on back of the unit under test.

(13) Set RX CAL Switch to the 31 position.
(14) Set variable power supply to 3.29 (±0.05) vdc.
(15) Verify that Null Meter is at the full scale indices ±1/2 needle width.

NOTE: View indicator from a minimum of two feet with eye at center of meter. Power supply voltage may be verified using (+) and (-) test points on back of the unit under test.

(16) Disconnect variable power supply.
(17) Jumper pins N, Z, F, K, E, F, and D together and connect them to (-) negative side of the variable power supply. Connect (+) positive side to pin E. Set variable power supply to 28 (±1.0) vdc. Verify the following lights are illuminated:
  (a) WARN light
  (b) PUSHER light
  (c) SQUAT light
  (d) 3° FLAP light
  (e) 13° FLAP light
  (f) 25° FLAP light
  (g) POWER light

NOTE: The SHAKE light shall not be illuminated.

(18) Remove all pin connections made in step (17).
(19) Connect (-) negative side of variable power supply to pin D and (+) positive side to pin T.
(20) Set variable power supply to 28 (±1.0) vdc and verify that SHAKE light is illuminated and all other lights are still extinguished. Disconnect variable power supply.
(21) Confirm diode integrity by using the ohmmeter positive and negative leads at the pin connections shown in Table 3.

NOTE: To establish a forward bias condition, the scale on the ohmmeter may have to be expanded until enough voltage/current is available to forward bias the diode. The diodes associated with pin D will only be forward biased.

(+) Lead	(-) Lead	Verify
Pin A	Pin N	Forward Bias
Pin N	Pin A	Back Bias
Pin E	Pin N	Forward Bias (plus light bulb resistance.)
Pin N	Pin E	Back Bias
Pin D	Box Gnd	Forward Bias
Box Gnd	Pin D	Forward Bias

Table 3

NOTE: Forward bias diode will measure approximately 15 ohms. Back bias diode will measure 100K ohms or greater. Light bulb resistance will vary depending on multimeter used (approximately 200 to 900 ohms).

(22) Connect (+) positive lead of ohmmeter to pin R and (-) negative lead to pin A. Verify the resistance value as shown for condition 1 of Table 4.
(23) Verify condition 2 by slowly adjusting the Vane Hi potentiometer CCW. Verify that the measured resistance value decreases in a smooth manner as it is adjusted. Verify the ohms value when fully adjusted CCW.
(24) Verify condition 3 by slowly adjusting the digital potentiometer CCW. Verify that the measured resistance value decreases in a smooth manner as it is adjusted. Verify the ohms value when fully adjusted CCW.
(25) Verify condition 4 by slowly adjusting the Vane Level potentiometer CCW. Verify that the measured resistance value decreases in a smooth manner as it is adjusted. Verify the ohms value when full adjustment CCW.
(26) Return Vane Level and Vane Hi potentiometers to approximate center of travel (11 turns).

Condition	Vane Hi Pot.	Vane Level Pot.	Digital Pot.	Measurement (Ohms)
1	CW	CW	CW	1330 (±150)
2	CCW	CW	CW	1106 (±130)
3	CCW	CW	CCW	776 (±100)
4	CCW	CCW	CCW	0 (+10)

Table 4

(27) Perform continuity check of stall warning ground calibration box wiring harness. (Refer to Table 5 for test points.)

NOTE: Plug P1 mates to the test set, plug P2 mates to the computer and receptacle J1 mates to aircraft harness.

Harness Wiring	
From	To
P1-A	J1-A
P1-D	J1-P
P1-E	J1-E
P1-F	J1-F
P1-G	J1-G
P1-H	J1-H
P1-K	J1-U
P1-L	J1-L
P1-M	J1-M
P1-N	P2-A
P1-P	P2-P
P1-T	P2-T
P1-U	P2-U
P1-X	P2-F
P1-Y	P2-G
P1-Z	P2-Z
P1-a	P2-a
P1-b	J1-b
P1-c	J1-c
P1-e	J1-e
P1-f	J1-f
P1-g	J1-a
P1-k	P2-E
P1-m	J2-g
P1-n	P2-H
P1-p	P2-L
P1-q	P2-M
P1-r	P2-b
P1-s	P2-c
P1-u	P2-e
P1-v	P2-f
P1-w	P2-h
P1-y	J1-g

Table 5

ANGLE-OF-ATTACK TRANSDUCER - MAINTENANCE PRACTICES

1. **GENERAL**
   A. The angle-of-attack transducers continuously monitor the aircraft wing lift loading and provide this information to the Stall Warning computer.
   B. An angle-of-attack transducer is located on each side of the aft nose compartment.

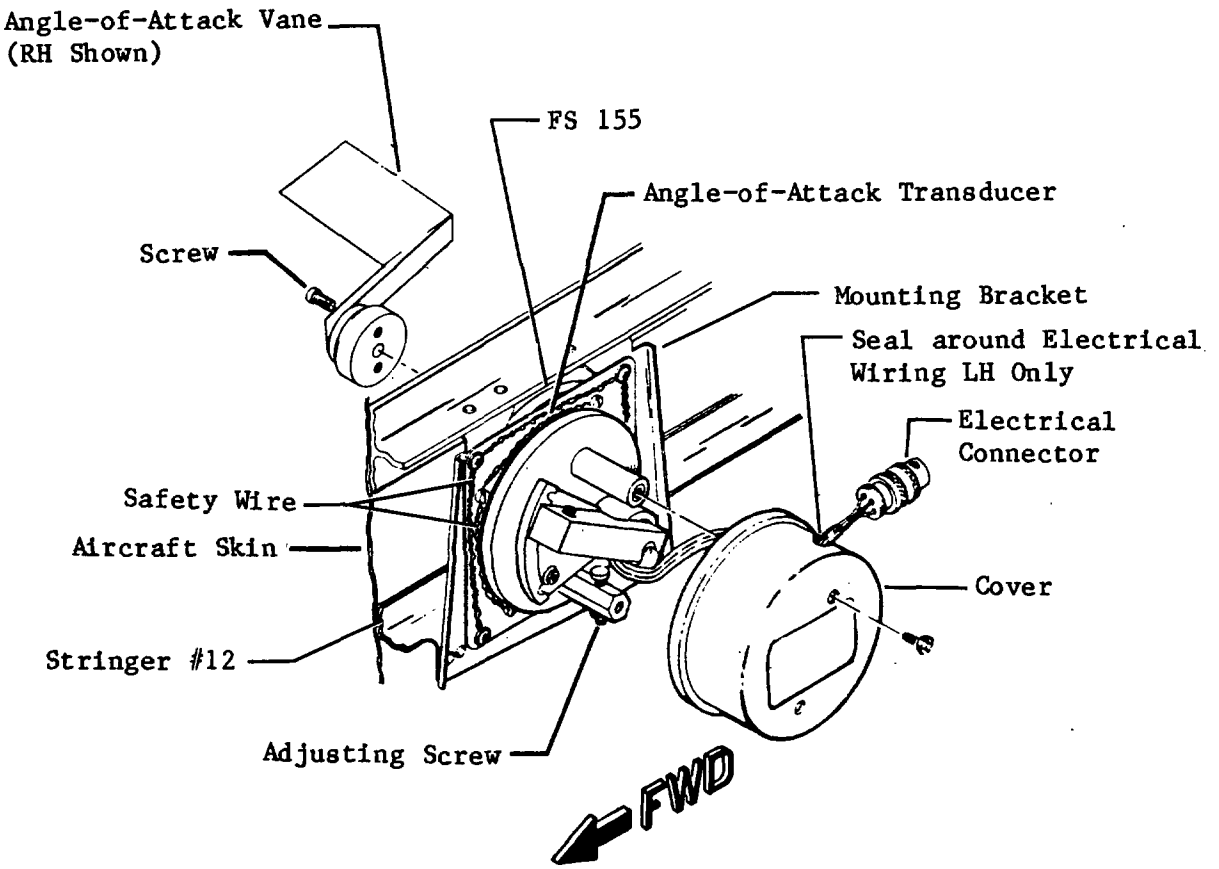
2. **REMOVAL/INSTALLATION**

CAUTION: IF ANGLE-OF-ATTACK TRANSDUCERS ARE REPLACED, A TEST AND ADJUSTMENT OF THE STALL WARNING ELECTRICAL COMPONENTS MUST BE PERFORMED. REFER TO 27-31-00.

NOTE: Removal and installation procedures are identical for both angle-of-attack transducers.

A. Removal Angle-of-Attack Transducer (See figure 201.)
   (1) Remove nose compartment access doors.
   (2) Remove screws securing vane to transducer. Remove vane from transducer.
   (3) Disconnect electrical connector from transducer.
   (4) Remove safety wire, attaching parts and mounting bracket with transducer attached from aircraft.

B. Install Angle-of-Attack Transducer (See figure 201.)
   (1) Install transducer; secure with attaching parts and safety wire.
   (2) Connect electrical connector to transducer.
   (3) Install vane on transducer and secure with attaching parts.
   (4) Install nose compartment access doors.
   (5) Perform Stall Warning Functional Tests and make adjustments as necessary. (Refer to Adjustment/Test in 27-31-00.)



RH Transducer Shown

Angle-of-Attack Transducer Installation
Figure 201

CONTROL COLUMN SHAKER - MAINTENANCE PRACTICES

1. **REMOVAL/INSTALLATION**

   A. Remove Control Column Shaker (See figure 201.)

   (1) Assure that Battery and Stall Warning Switches are OFF.
   (2) Disconnect electrical wiring from control column disconnect connector.
   (3) Remove attaching parts and shaker from control column.

   B. Install Control Column Shaker (See figure 201.)

   (1) Install shaker on control column and secure with attaching parts.
   (2) Install shrink tubing (P/N RNF 100 type T) over wiring.
   (3) Crimp new pins (P/N 66107-1) to wires and insert wires into electrical connector.
   (4) Using an ohmmeter, perform continuity check as follows:
      (a) Gain access to the stall warning computer by removing the nose compartment access doors.
      (b) On Aircraft 35-002 thru 35-066, 36-002 thru 36-017 (not equipped with Reduced Approach Speed System), disconnect plug P85 and verify continuity between E and ground and pin V and ground.
      (c) On Aircraft 35-067 and Subsequent, 36-018 and Subsequent, and earlier Aircraft equipped with Reduced Approach Speed System, disconnect plugs P913 and P914 and verify continuity between pins G and T on each plug.

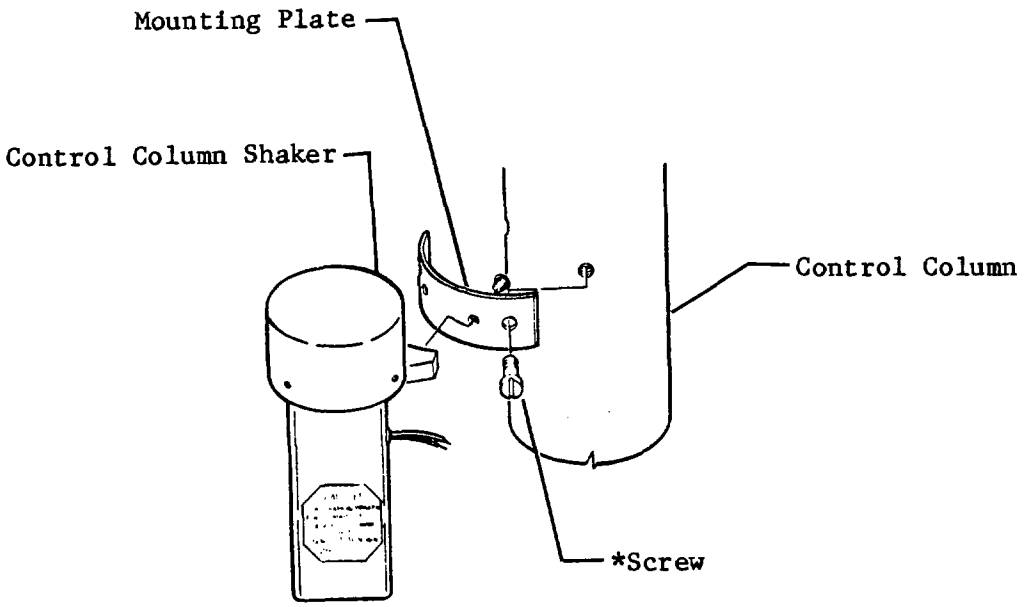
2. **ADJUSTMENT/TEST**

   A. Functional Test of Control Column Shaker (See figure 202.)

   NOTE: ° This functional test is to be performed in accordance with current inspection intervals in Chapter 5.

   ° The purpose of this functional test is to verify the control column shaker performance by measuring frequency.

   (1) Fabricate a tachometer assembly as shown in figure 202.
   (2) Check shaker mounting screws for tightness. Looseness will "unload" the shaker, change the frequency, and reduce the force transmitted.
   (3) Trim control column to mid-travel so that elevator stops are not contacted and little or no manual force is required to hold the column upright during this procedure.
   (4) Tape tachometer to LH control wheel as shown in figure 202 so that straight end is free to vibrate.
   (5) Set Battery Switch(es) and LH Stall Warning Switch to ON.
   (6) Raise LH stall vane far enough to get a steady state shaker (no pulsing), but not far enough to engage pusher.
   (7) Holding control wheel firmly with one hand, adjust exposed length of wire for maximum resonance.



*Use correct length screw to prevent interference with control column cables.

Control Column Shaker Installation
Figure 201

NOTE: Resonance is easily recognized when the end of the wire vibrates with an amplitude of one inch or more. Adjust length of wire as required to produce the maximum amplitude of vibration at tip of wire. Ensure that the control column is not on stops or touching anything except operator's hand during this operation, since any such contact could change frequency.

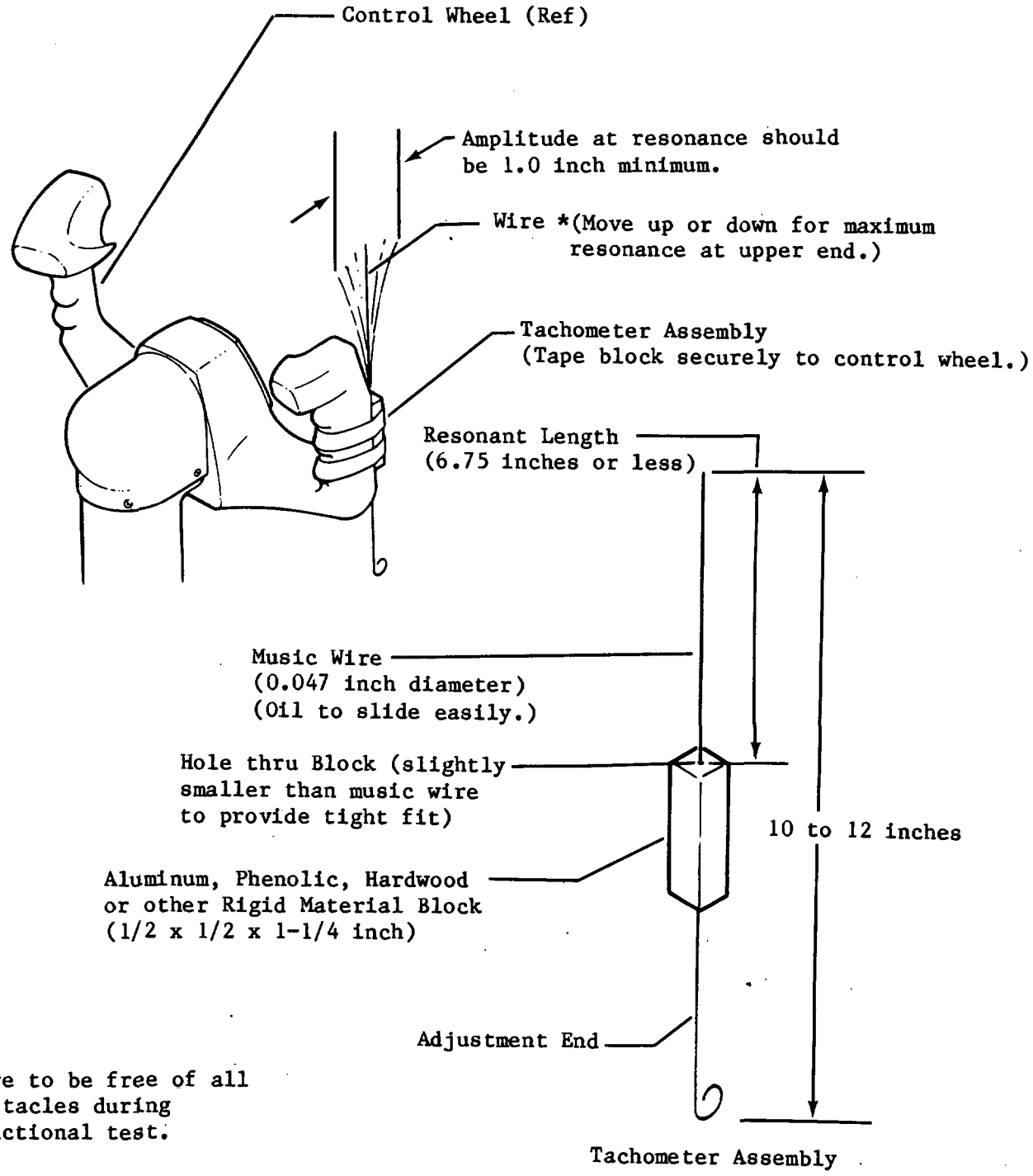
(8) Lower LH stall vane, stopping shaker operation.

CAUTION: DO NOT DISTURB WIRE LENGTH.

(9) Carefully measure resonant length of wire. Wire length should be no longer than 6.75 inches.

(10) If the resonant length of wire is greater than 6.75 inches, there is a problem in the system. Longer resonant length means too low a frequency, which may be caused by a weak motor, low input voltage, or loose mounting screws.

(11) Perform steps A.(2) thru A.(10) using the RH control wheel, RH Stall Warning Switch, and RH stall vane.

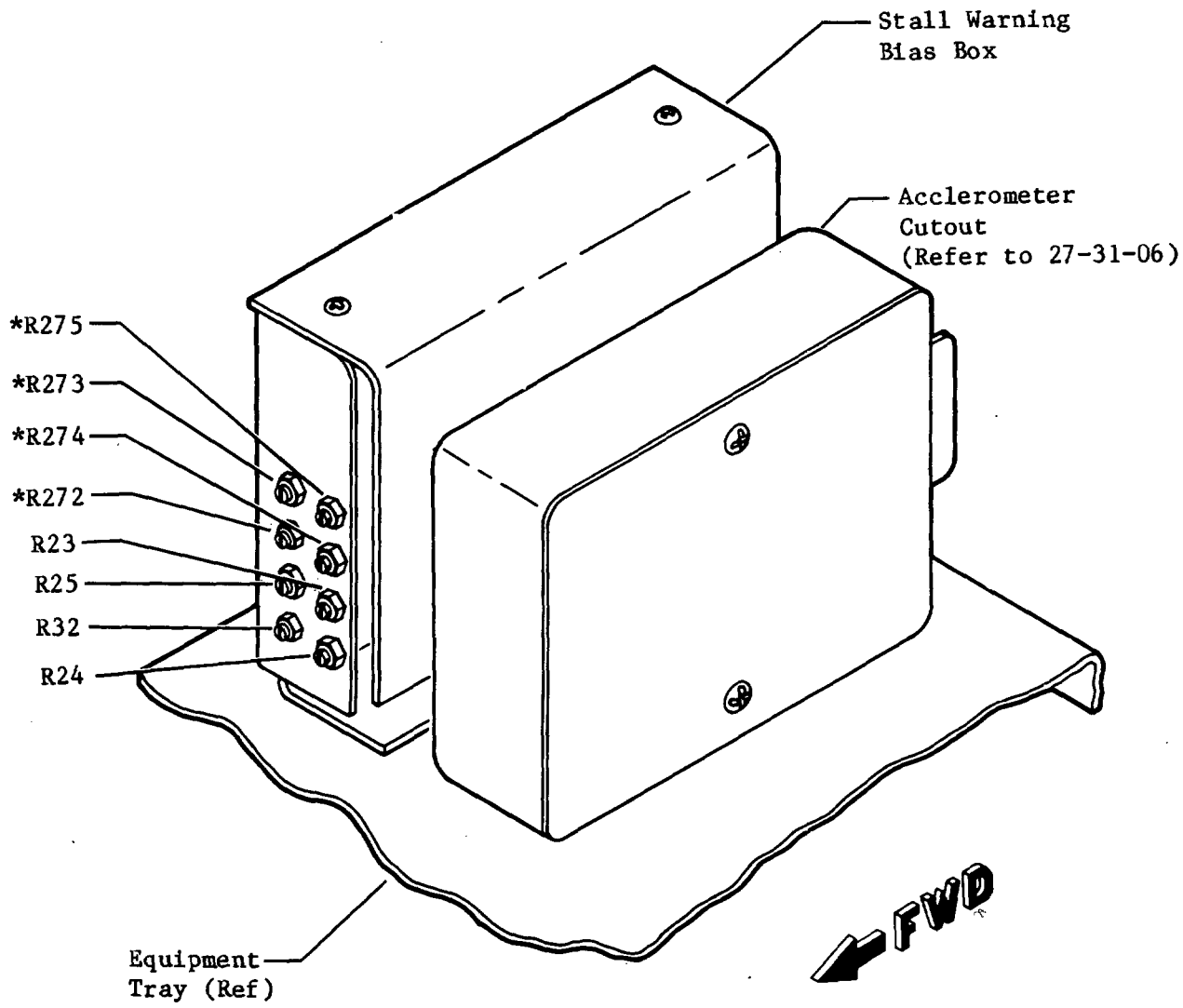


Tachometer Assembly
Figure 202

STALL WARNING BIAS BOX - MAINTENANCE PRACTICES

1. **REMOVAL/INSTALLATION**
   - **A. Remove Bias Box** (See figure 201.)
     - (1) Remove nose compartment access doors.
     - (2) Disconnect electrical connector from bias box.
     - (3) Remove attaching parts and bias box from aircraft.
   - **B. Install Bias Box** (See figure 201.)
     - (1) Install bias box and secure with attaching parts.
     - (2) Connect electrical connector to bias box.
     - (3) Install nose compartment access doors.
     - (4) Perform the Stall Warning Functional Tests and make adjustments as necessary. (Refer to Adjustment/Test in 27-31-00.)

*Potentiometer Effective Aircraft
35-062 thru 35-066



Stall Warning Bias Box Installation
Figure 201

9-102B

STALL WARNING COMPUTER-AMPLIFIER - MAINTENANCE PRACTICES

1. **REMOVAL/INSTALLATION**

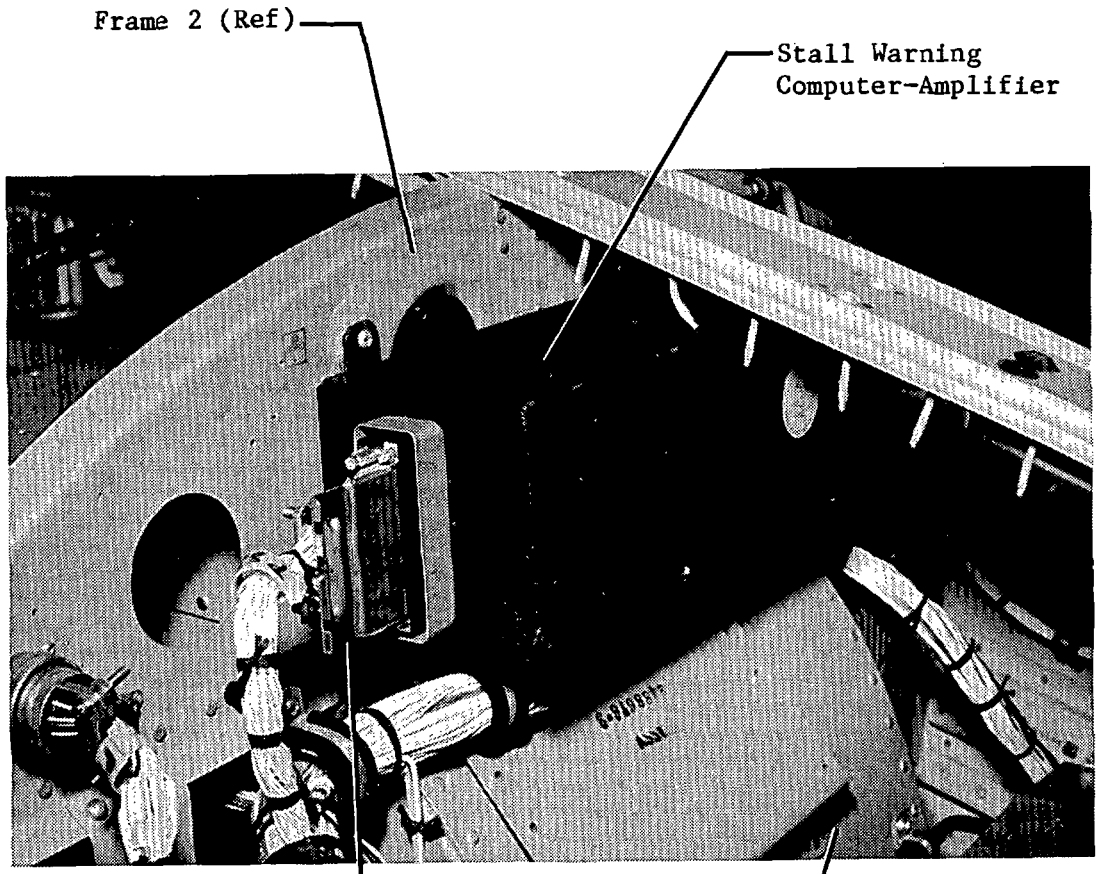
   A. Remove Computer-Amplifier (See figures 201 thru 203.)

   (1) Remove nose compartment access doors.
   (2) Disconnect electrical connectors from computer-amplifier.
   (3) Remove attaching parts and computer-amplifier from aircraft.

   B. Install Computer-Amplifier (See figures 201 thru 203.)

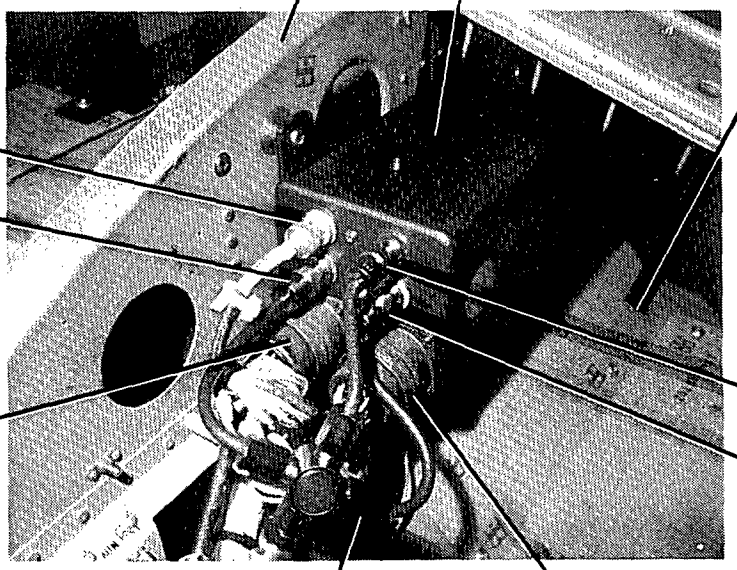
   CAUTION: ON AIRCRAFT 35-067 AND SUBSEQUENT, 36-018 AND SUBSEQUENT, AND EARLIER AIRCRAFT EQUIPPED WITH REDUCED APPROACH SPEED SYSTEM, MINIMAL FORCE SHOULD BE USED WHEN CONNECTING ELECTRICAL CONNECTORS (P913 & P914) TO COMPUTER. CONNECTORS ARE CLOCKED TO PREVENT CONNECTION TO WRONG RECEPTACLES; HOWEVER, CONNECTORS CAN BE FORCED ON WRONG RECEPTACLE. DAMAGE TO COMPUTER-AMPLIFIER WILL RESULT IF CONNECTORS ARE CONNECTED INCORRECTLY.

   (1) Install computer-amplifier and secure with attaching parts.
   (2) Connect electrical connectors to computer-amplifier.
   (3) Install nose compartment access doors.
   (4) Perform 1/2 g accelerometer functional test. (Refer to 27-31-05.)
   (5) Perform Stall Warning Functional Tests and make adjustments as necessary. (Refer to Adjustment/Test in 27-31-00.)



ORIGINAL
As Received By
ATP

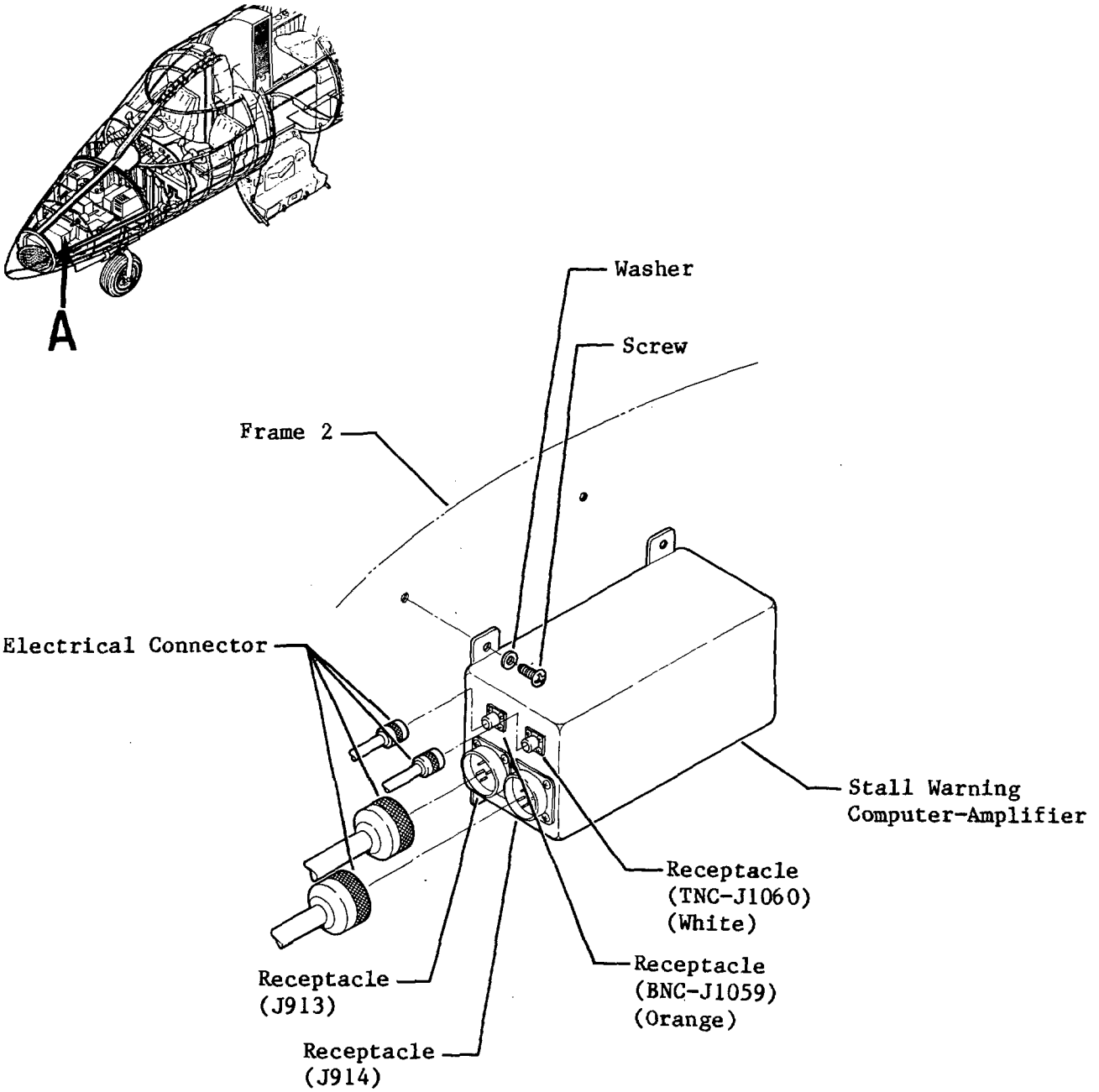
Stall Warning Computer-Amplifier Installation
Figure 201



ORIGINAL
As Received By ATP

CAUTION: CONSULT WIRING MANUAL TO ENSURE PROPER PLUG MATING TO COMPUTER-AMPLIFIER RECEPTACLE.

Stall Warning Computer-Amplifier Installation
Figure 202



Detail A

Stall Warning Computer-Amplifier Installation
Figure 203

14-148A-2

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Multimeter	Model 8012A	John J. Fluke Co. Everett, WA	Check resistances.

2. Removal/Installation

NOTE: The accelerometer is installed on the aft LH side of frame 2. A hole in frame 2 allows the accelerometer to protrude through the frame.

A. Remove Accelerometer. (See Figure 201.)
   (1) Remove electrical power from aircraft.
   (2) Remove LH nose compartment access door.
   (3) Disconnect electrical connector from accelerometer.
   (4) Remove attaching parts and accelerometer from aircraft.

B. Install Accelerometer. (See Figure 201.)
   (1) Install accelerometer, ensuring that vertical axis of accelerometer is in the vertical position, and secure with attaching parts.
   (2) Connect electrical connector to accelerometer.
   (3) Install LH nose compartment access door.
   (4) Restore electrical power to aircraft.

3. Adjustment/Test

A. Adjust Accelerometer. (*Effective Aircraft equipped with P/N 2380071 Accelerometer*)
   (1) Remove accelerometer from aircraft.
   (2) Place accelerometer on bench in its normal mounting position.
   (3) Check resistance between pins A and B and E and F of connector.
   (4) Resistance shall be less than 20 ohms in each check.
   (5) If resistance is greater than 20 ohms, remove cover from accelerometer.
   (6) Add washers (P/N AN960-3L) under each stop plate screw between housing base and upper copper arm to position accelerometer mass downward. Use as many washers as needed to obtain a resistance of less than 20 ohms.
   (7) Recheck resistance between pins A and B and E and F. Resistance shall be less than 20 ohms.
   (8) Install cover on accelerometer.
   (9) Connect meter to pins A and B. With accelerometer in normal mounting position, rotate accelerometer 40° (±10°) CW and CCW.
   (10) The resistance reading shall not increase. Beyond 40° (±10°) CW or CCW, the resistance reading shall increase smoothly.
   (11) Pitch the accelerometer 40° (±10°) forward and aft.
   (12) The resistance reading shall not increase. Beyond 40° (±10°) forward and aft, the resistance reading shall increase smoothly.
   (13) Repeat steps (9) thru (12) with meter connected to pins E and F.

B. Functional Test. (*Effective Aircraft equipped with P/N 6608283-2 Accelerometer*)
   (1) Remove accelerometer from aircraft.
   (2) Place accelerometer on bench in its normal mounting position.
   (3) Check resistance between pins A and E of connector.
   (4) Resistance shall be less than 0.4 ohm.

(5) If resistance is more than 0.4 ohm, ensure that accelerometer is on a flat, level surface and in the vertical plane.
(6) Rotate accelerometer forward until meter reads infinity. Measure the angle from vertical to the point where meter reads infinity. (Accelerometer point of reference is the mounting plate.) The angle shall be between 53° and 63.5°.

NOTE: If the angle where meter indicates infinity is not within tolerance, repeat test ensuring that accelerometer is in the horizontal plane.

(7) Repeat step (6), rotating accelerometer aft. Return accelerometer to its normal mounting position.
(8) Rotate accelerometer to the right until meter reads infinity. Measure the angle from horizontal to the point where meter reads infinity. (Accelerometer point of reference is bottom edge of mounting plate.) The angle shall be between 53° and 63.5°.

NOTE: If the angle where meter indicates infinity is not within tolerance, repeat test ensuring that accelerometer is in the vertical plane.

(9) Repeat step (8) rotating accelerometer left.
(10) If accelerometer does not function as described, it must be replaced.

C. Functional Test. (*Effective Aircraft equipped with P/N 6608283-1 Accelerometer*)
(1) Remove accelerometer from aircraft.
(2) Place accelerometer on bench in its normal mounting position.
(3) Check resistance between pins B and C and E and F of connector.
(4) Resistance shall be less than 0.2 ohm.
(5) If resistance is more than 0.2 ohm, ensure that accelerometer is on a flat, level surface and in the vertical plane.
(6) Attach meter to pins B and C. Rotate accelerometer forward until meter reads infinity. Measure the angle from vertical to the point where meter reads infinity. (Accelerometer point of reference is the mounting plate.) The angle shall be between 53° and 63.5°.

NOTE: If the angle where meter reads infinity is not within tolerance, repeat test ensuring that accelerometer is in the horizontal plane.

(7) Repeat step (6) rotating accelerometer aft. Return accelerometer to its normal mounting position.
(8) Rotate accelerometer to the right until meter reads infinity. Measure the angle from horizontal to the point where meter reads infinity. (Accelerometer point of reference is bottom edge of mounting plate.) The angle shall be between 53° and 63.5°.

NOTE: If the angle where meter indicates infinity is not within tolerance, repeat test ensuring that accelerometer is in the vertical plane.

(9) Return accelerometer to its normal mounting position. Repeat step (8) rotating accelerometer left.
(10) Repeat steps (6) through (8) with meter attached to pins E and F of connector.
(11) If the accelerometer does not function as described, it must be replaced.

4. Inspection/Check

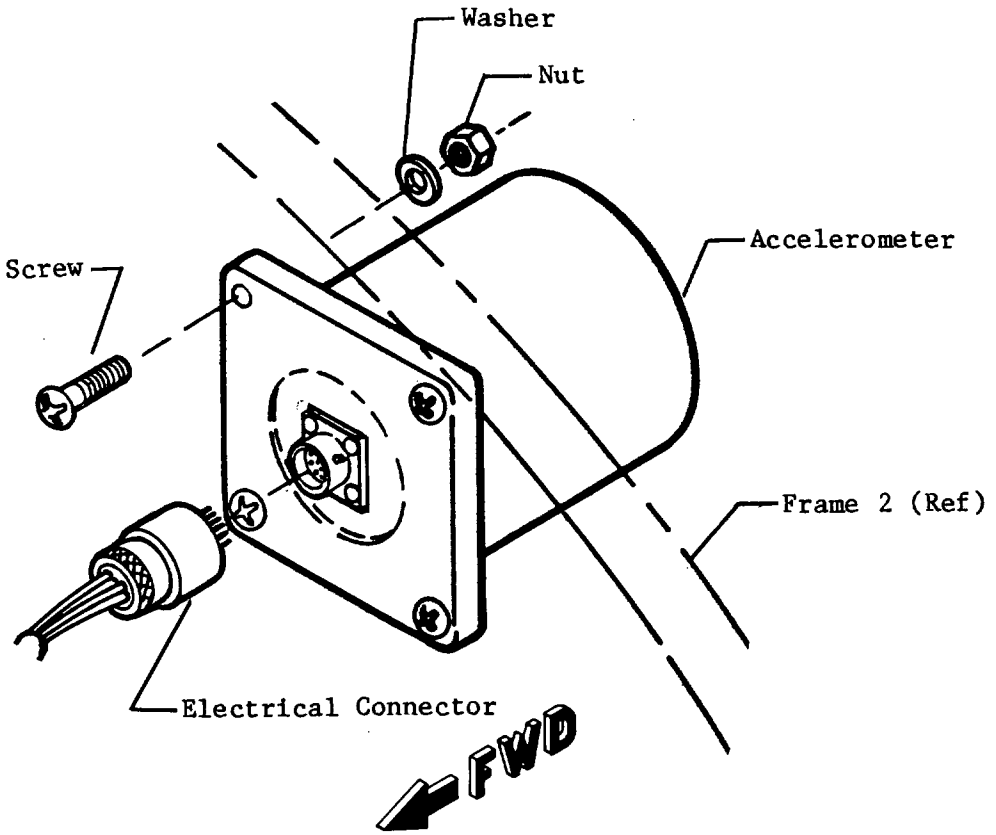
A. Operational Check of Stall Warning Accelerometer. (See Figure 201.)

NOTE:
- Perform operational check of stall warning accelerometer in accordance with current inspection intervals specified in Chapter 5.
- Performance of this operational check is in compliance with AD 82-01-05.

(1) Remove LH nose compartment access door.
(2) Remove accelerometer attaching parts and have one person hold accelerometer in its normal position.
(3) Connect external power to aircraft and set Battery Switches on. Ensure that 28 (±0.5) vdc is available.

CAUTION: LIMIT TIME OF THIS OPERATIONAL CHECK TO PREVENT OVERHEATING COMPONENTS.

(4) Set L Stall Warning Switch on.
(5) Adjust horizontal stabilizer position until a neutral feel of control column is obtained.
(6) Attach a spring scale (0 to 100 pounds) to control column.
(7) Hold accelerometer in its normal position. Move the L angle-of-attack vane up slowly until pusher actuates. Verify pusher force at control column is 40 to 80 pounds (18.14 to 36.29 kilograms) .
(8) Rotate accelerometer 180° (upside down from normal position).
(9) Check that force at control column is reduced to 35 pounds (15.87 kilograms) or less.
(10) Return accelerometer to normal position and set L Stall Warning Switch off.
(11) Repeat steps (1) thru (10) using R Stall Warning Switch and R angle-of-attack vane.
(12) If accelerometer fails operational check, perform functional test and repair or replace as required.
(13) Set Battery Switches and Stall Warning Switches off and disconnect external power.
(14) Install LH nose compartment access door.



Accelerometer Installation
Figure 201

STALL WARNING ACCELEROMETER CUTOUT - MAINTENANCE PRACTICES

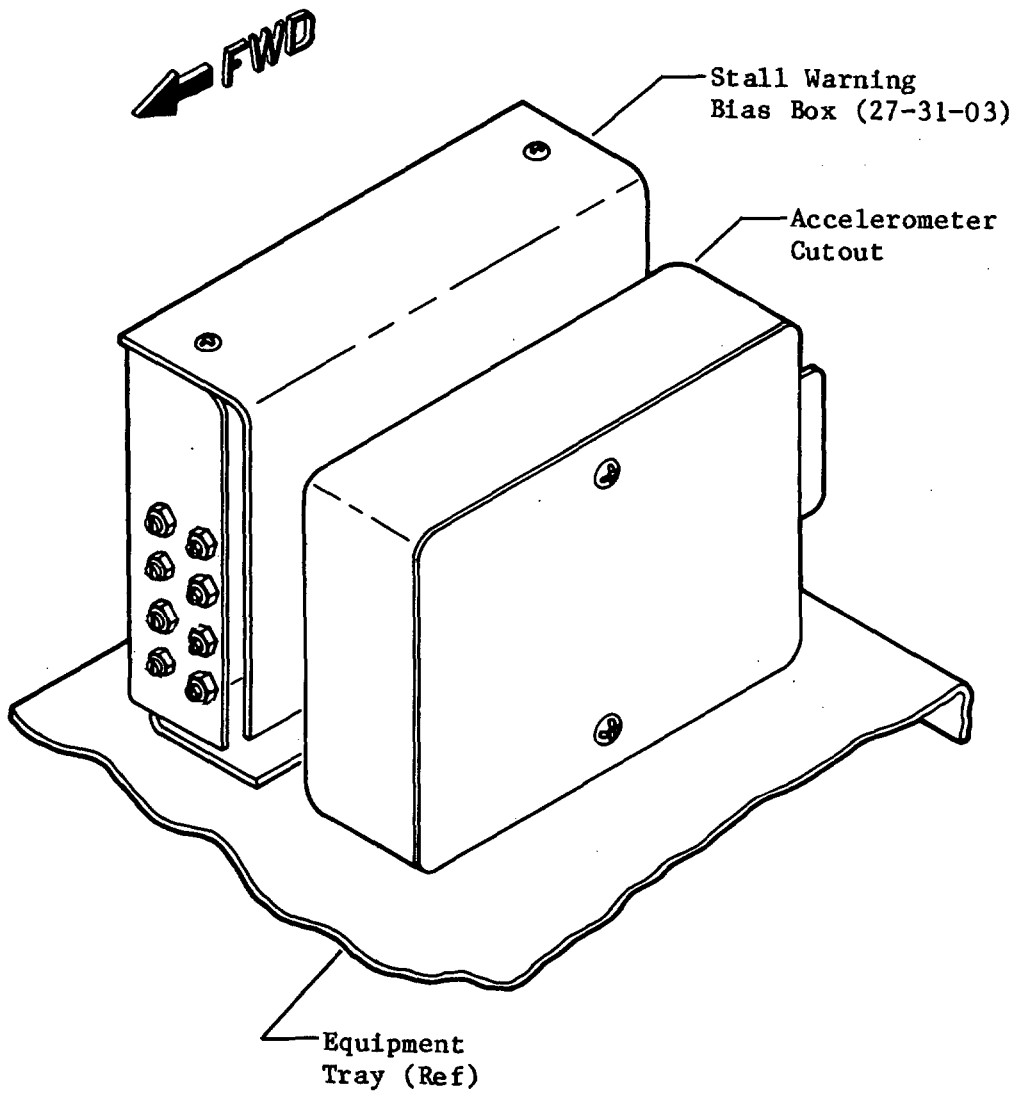
1. REMOVAL/INSTALLATION

A. Remove Accelerometer Cutout (See figure 201.)

(1) Remove nose compartment access doors.
(2) Disconnect electrical connector from accelerometer cutout.
(3) Remove attaching parts and accelerometer cutout from aircraft.

B. Install Accelerometer Cutout (See figure 201.)

(1) Install accelerometer cutout and secure with attaching parts.
(2) Connect electrical connector to accelerometer cutout.
(3) Install nose compartment access doors.



Stall Warning Accelerometer Cutout Installation
Figure 201

STALL WARNING ALTITUDE SWITCH - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Altitude Pressure Switch (S367, LH; S366, RH) (See Figure 201.)
   (1) Remove nose compartment access doors.
   (2) Gain access to applicable altitude pressure switch.
   (3) Disconnect electrical connector (P53, LH; P54, RH) from altitude pressure switch.
   (4) Loosen and remove altitude pressure switch from static lines. Cap all exposed fittings.

B. Install Altitude Pressure Switch (S367, LH; S366, RH) (See Figure 201.)
   (1) Remove caps and install and secure altitude pressure switch to static line.
   (2) Connect electrical connector (P53, LH; P54, RH) to altitude pressure switch.
   (3) Perform pitot system leakage check. (Refer to Chapter 34.)
   (4) Perform operational check of altitude pressure switch. (Refer to Inspection/Check, this section.)
   (5) Install nose compartment access door.
   (6) Restore aircraft to normal.

2. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Pitot Static Test Set	1811F	Barfield Instrument Co. Atlanta, GA	Check switch.

3. Inspection/Check

NOTE: The altitude pressure switch is a part of the airspeed/mach indicator on *Aircraft 35-506 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot*." (Refer to Chapter 34 for operational check.)

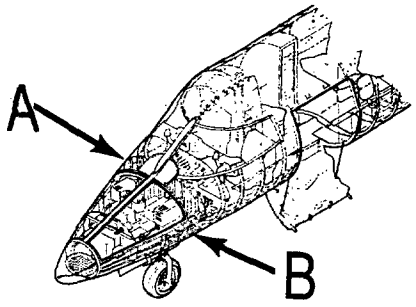
A. Operational Check of Altitude Pressure Switches (S367, LH; S366, RH)

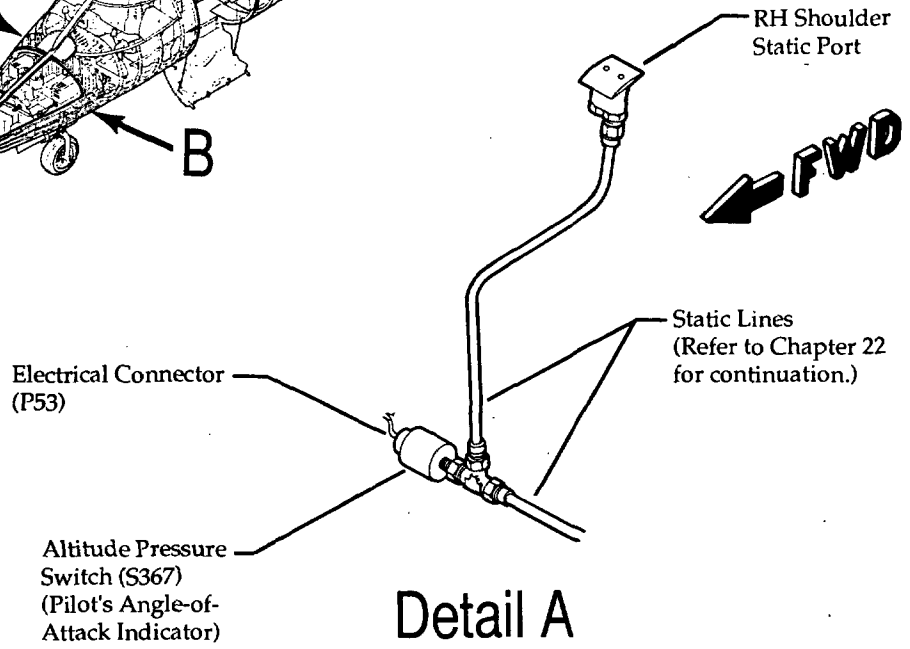
NOTE: Perform operational check of altitude pressure switches in accordance with current inspection interval specified in Chapter 5.

(1) Set Battery, Primary and Secondary Inverter, L and R Stall Warning Switches on.
(2) Lower both angle-of-attack vanes so that angle-of-attack indicator is midway through the green band.
(3) Position flaps to full up.

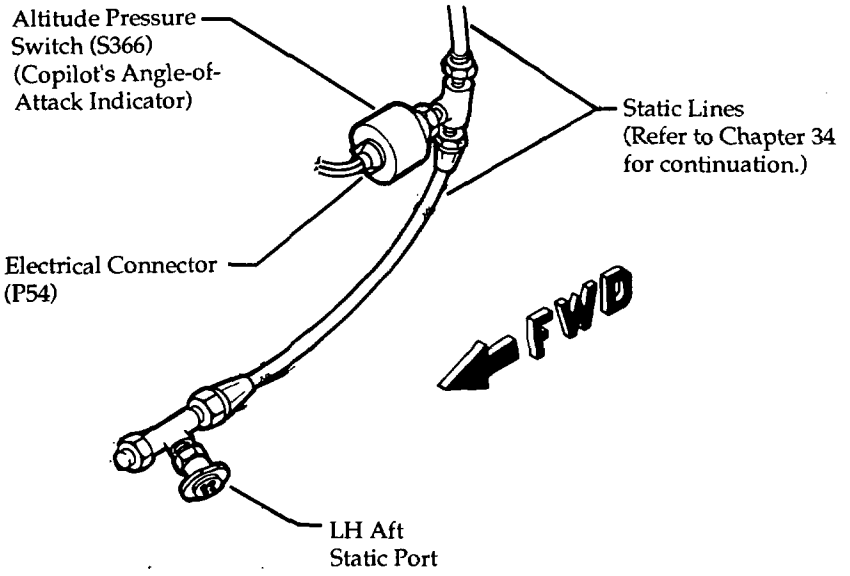
CAUTION: TO AVOID DAMAGE TO AIRCRAFT INSTRUMENTS, DO NOT EXCEED 5000 FEET PER MINUTE RATE OF CLIMB OR DESCENT. DO NOT ALLOW AIRSPEED TO DECREASE BELOW ZERO.

(4) Connect pitot static test set to shoulder static ports and slowly apply vacuum.
(5) Observe pilot's angle-of-attack indicator and tester altimeter. Pilot's angle-of-attack indicator shall abruptly move toward yellow area (stall) when tester altimeter reads 22,500 (±750) feet.





Detail A



Detail B

A9-24C-3
B14-28C-11

Stall Warning Altitude Switch Installation
Figure 201

EFFECTIVITY: 35-067 THRU 35-505, AND 36-018 THRU 36-053 AND PRIOR AIRCRAFT EQUIPPED WITH REDUCED APPROACH SPEED SYSTEM, BUT NOT MODIFIED PER AAK 83-2, "INSTALLATION OF FC-530 AUTOPILOT"

CAUTION:
- **TO AVOID DAMAGE TO AIRCRAFT INSTRUMENTS, DO NOT EXCEED 5000 FEET PER MINUTE RATE OF CLIMB OR DESCENT. DO NOT ALLOW AIRSPEED TO DECREASE BELOW ZERO.**
- **TO AVOID DAMAGE TO AIRCRAFT INSTRUMENTS, DO NOT EXCEED 20 KNOTS PER SECOND WHEN VENTING ATMOSPHERIC PRESSURE INTO PITOT SYSTEM.**

(6) Slowly bleed off vacuum until field elevation is reached. Remove tester from shoulder ports.

CAUTION: **TO AVOID DAMAGE TO AIRCRAFT INSTRUMENTS, DO NOT EXCEED 5000 FEET PER MINUTE RATE OF CLIMB OR DESCENT. DO NOT ALLOW AIRSPEED TO DECREASE BELOW ZERO.**

(7) Connect tester to RH forward and LH aft static ports and slowly apply vacuum.
(8) Observe copilot's angle-of-attack indicator and tester altimeter. Copilot's angle-of-attack indicator shall abruptly move toward yellow area (stall) when tester altimeter reads 22,500 (±750) feet.

CAUTION:
- **TO AVOID DAMAGE TO AIRCRAFT INSTRUMENTS, DO NOT EXCEED 5000 FEET PER MINUTE RATE OF CLIMB OR DESCENT. DO NOT ALLOW AIRSPEED TO DECREASE BELOW ZERO.**
- **TO AVOID DAMAGE TO AIRCRAFT INSTRUMENTS, DO NOT EXCEED 20 KNOTS PER SECOND WHEN VENTING ATMOSPHERIC PRESSURE INTO PITOT SYSTEM.**

(9) Slowly bleed off vacuum until field elevation is reached. Remove tester from aircraft.
(10) Set Battery, Primary and Secondary Inverter, L and R Stall Warning Switches off.
(11) Restore aircraft to normal.

ANGLE-OF-ATTACK VANE - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

WARNING: AVOID CONTACT WITH HOT ANGLE-OF-ATTACK VANES.

NOTE: ° Removal and installation procedures for both angle-of-attack vanes are identical.

° Replacement of the stall transducer vane is permitted in the event of a failed heater. Physical damage to the vane requires total transducer replacement because of possible damage to the transducer shaft, bearing, and potentiometer. Vane replacement requires functional test to verify that shaker and pusher actuation occurs within the prescribed tolerance of the vane plate marks.

A. Remove Angle-of-Attack Vane (See figure 201.)

(1) Remove electrical power from aircraft.

(2) Ensure that angle-of-attack vanes are not hot.

(3) Remove attaching parts securing angle-of-attack vane to angle-of-attack transducer assembly.

CAUTION: ° DO NOT PRY ANGLE-OF-ATTACK VANE OFF, OR ALLOW IT TO COCK TO ONE SIDE WHILE REMOVING AS DAMAGE TO THE TRANSDUCER ASSEMBLY COULD RESULT.

° A SLIGHT FORWARD AND AFT MOTION WHILE APPLYING PRESSURE TO VANE WILL DISENGAGE VANE FROM TRANSDUCER ASSEMBLY.

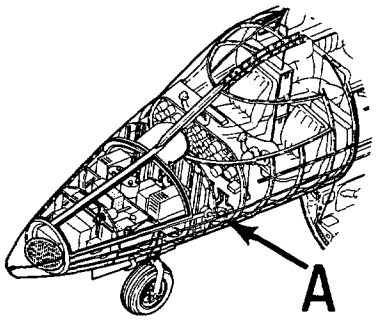
(4) Carefully remove vane from transducer assembly.

B. Install Angle-of-Attack Vane (See figure 201.)

(1) Position angle-of-attack vane at its appropriate location on angle-of-attack transducer assembly, and secure with attaching parts. Alternately tighten screws until vane is seated firmly in place.

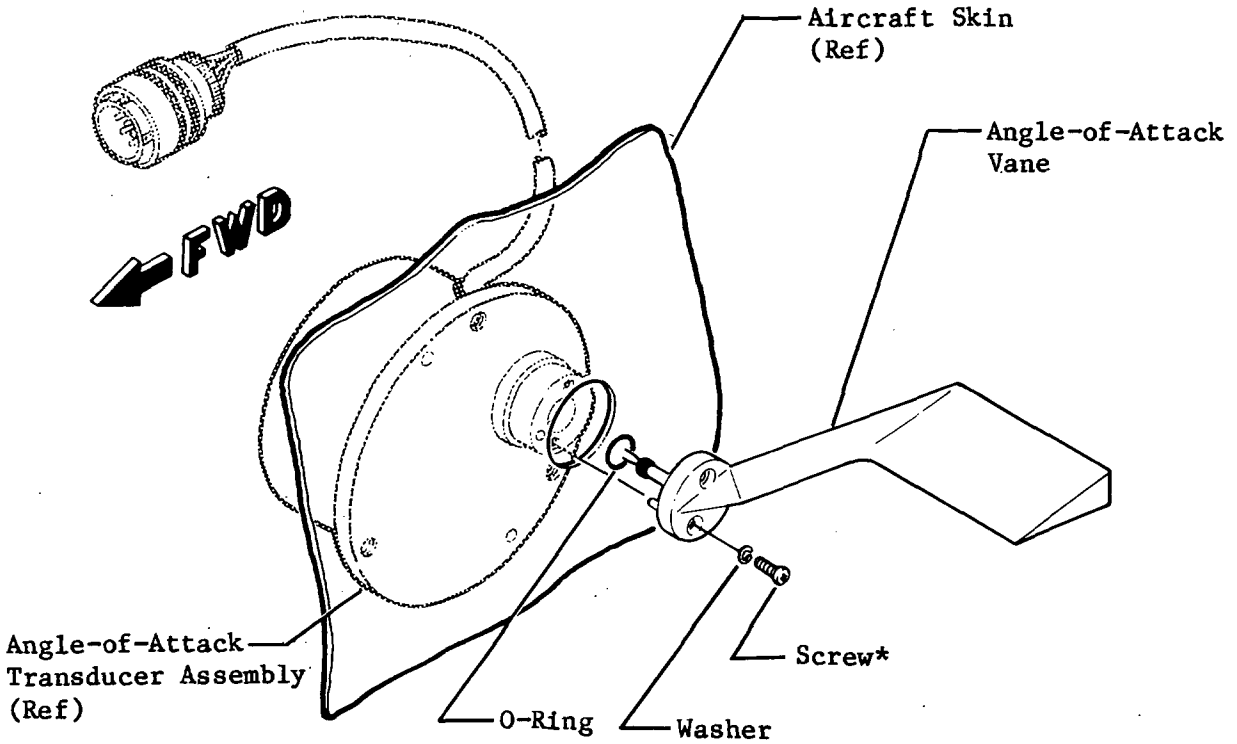
(2) Restore electrical power to aircraft. Perform steps 1.D.(1), 1.D.(2) and 1.D.(11) thru 1.D.(26) as applicable of the stall warning system functional test.

(3) Restore aircraft to normal.



CAUTION: ° DO NOT PRY ANGLE-OF-ATTACK VANE OFF, OR ALLOW IT TO COCK TO ONE SIDE WHILE REMOVING AS DAMAGE TO THE TRANSDUCER ASSEMBLY COULD RESULT.

° A SLIGHT FORWARD AND AFT MOTION WHILE APPLYING PRESSURE TO VANE WILL DISENGAGE VANE FROM TRANSDUCER ASSEMBLY.



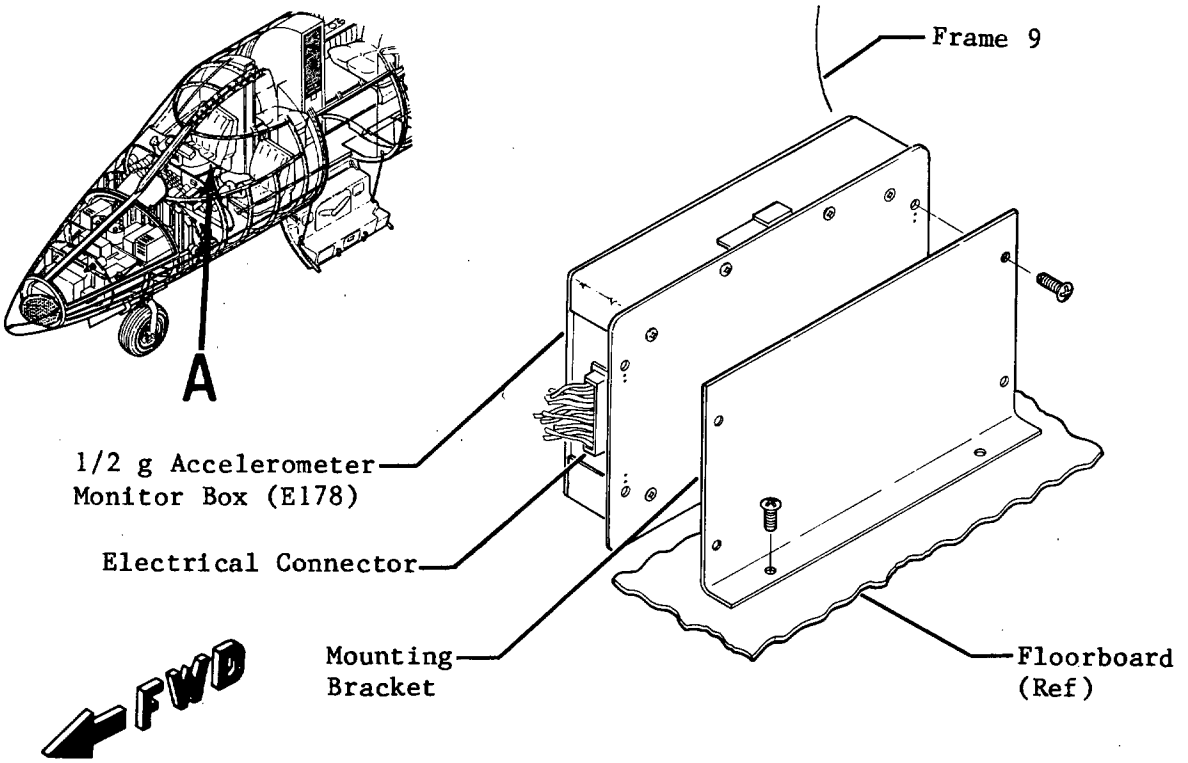
*Torque 10-14 inch-pounds

Detail A

Angle-of-Attack Vane Installation
Figure 201

1/2 g ACCELEROMETER MONITOR BOX - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Remove 1/2 g Accelerometer Monitor Box. (See Figure 201.)
      (1) Remove electrical power from aircraft.
      (2) Gain access to 1/2 g accelerometer monitor box, located beneath the copilot's seat, forward of frame 9 (FS 200) at RBL 12.
      (3) Disconnect electrical connector from 1/2 g accelerometer monitor box.
      (4) Remove attaching parts and remove 1/2 g accelerometer monitor box from mounting bracket.
   B. Install 1/2 g Accelerometer Monitor Box. (See Figure 201.)
      (1) Position 1/2 g accelerometer monitor box on mounting bracket and secure with attaching parts.
      (2) Connect electrical connector to 1/2 g accelerometer monitor box.
      (3) Restore electrical power to aircraft.
      (4) Perform stall warning functional check. (Refer to 27-31-00.)



Detail A

9-293A

1/2 g Accelerometer Monitor Box
Figure 201

2. **Adjustment/Test**
   A. Functional Test of 1/2 g Accelerometer Monitor Box.

      NOTE: • Perform functional test of 1/2 g accelerometer box in accordance with current inspection intervals specified in Chapter 5.

      (1) Remove 1/2 g accelerometer monitor box from mounting bracket, leaving electrical connectors connected.
      (2) Have one person hold monitor box in its normal position.
      (3) Set Battery Switches and L and R Stall Warning Switches on.

      WARNING: AVOID CONTACT WITH HOT ANGLE-OF-ATTACK VANES.

      (4) Raise L angle-of-attack vane until pusher actuates.
      (5) With pusher actuated, turn 1/2 g accelerometer monitor box over. Pusher shall cease and nudger shall actuate with a pulsating force. A steady nudger force is not acceptable.

      NOTE: • If one system is commanding pusher and the remaining system is not at shaker, stick nudger force shall replace pusher force.

      • When the remaining system commands shaker (needle at intersection of green-yellow band), stick force will be at pusher level when systems warning annunciator is flashing and will revert to nudger when systems warning annunciator is not flashing.

      (6) Raise R angle-of-attack vane to shaker point. Pusher shall actuate with a pulsating force. Continue raising R vane. Force shall become steady.
      (7) Repeat steps (4) through (6) with R angle-of-attack vane at pusher and L angle-of-attack vane at shaker.
      (8) Install 1/2 g accelerometer monitor box and secure with attaching hardware.

3. **Approved Repairs**
   A. Replacement of Accelerometer Switches and Printed Circuit Boards.
      (1) Remove 1/2 g Accelerometer Monitor Box. (Refer to Removal/Installation, this section.)
      (2) Remove cover from box.
      (3) Remove screws securing Accelerometer Switch or Printed Circuit Board.

      CAUTION: DO NOT USE MAGNETIC FASTENERS TO ATTACH ACCELEROMETER SWITCHES. MAGNETIC FASTENERS WILL AFFECT SWITCH OPERATION.

      (4) Secure Accelerometer or Printed Circuit Board with attaching parts.
      (5) Secure cover with attaching parts.
      (6) Install monitor box. (Refer to Removal/Installation, this section.)

STALL WARNING VANE PLATE - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Inclinometer	2370002	Learjet Corp. Wichita, KS	Calibration.
Mylar Template	5471102-3 (LH)* 5471102-7 (LH)**	Learjet Corp. Wichita, KS	Calibration.
Mylar Template	5471102-4 (RH)* 5471102-8 (RH)**	Learjet Corp. Wichita, KS	Calibration.
MethylPropyl Ketone (MPK)		Commercially Available	Cleaning.
Masking Tape		Commercially Available	Surface protection.
Epoxy Adhesive	EC2216A (grey) EC2216B (white)	3M Co. St. Paul, MN	Plate installation.
Scribe		Commercially Available	Marking plate.
Clean Cotton Cloth		Commercially Available	Cleaning.

* *Aircraft 35-279 thru 35-505, and 36-045 thru 36-053 and prior aircraft equipped with stall fence (AAK 79-10 or AMK 83-5).*

** *Aircraft 35-408. 35-506 and Subsequent, and 36-053 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot."*

B. Removal of Stall Warning Vane Plate (See Figure 201.)

CAUTION: USE CARE WHEN REMOVING STALL WARNING VANE PLATE TO PREVENT DAMAGE TO AIRCRAFT SKIN.

(1) Remove screws from vane plate.
(2) Pry vane plate from aircraft using care not to damage aircraft skin.

C. Installation of Stall Warning Vane plate (See Figure 201.)
(1) Mask off vane plate mating surface.
(2) Thoroughly clean surfaces to be cemented using MPK. Wipe surface dry with a clean cotton cloth.
(3) Mix 100 parts, by weight, of EC-2216B (white) to 140 parts, by weight, of EC-2216A (gray). Thoroughly mix components until completely blended into a uniform gray paste.

(4) Apply adhesive 0.002 inch (0.051 mm) minimum thickness over both mating surfaces using a spatula. Place new vane plate over adhesive surface and secure with existing screws. Allow adhesive to cure 24 hours at room temperature. An accelerated full cure can be attained by heating for two hours at 180° F (82° C) or one hour at 250° F (121° C).

2. **Adjustment/Test**

A. Mark New Vane Plates (*Aircraft 35-002 thru 35-278 and 36-002 thru 36-044 not equipped with Stall Fence (AAK 79-10 or AMK 83-5)*).

NOTE: If stall warning pusher and shaker actuation is not within tolerance specified in 27-31-00 Maintenance Practices, paragraph 2.C or 2.F. replace vane plate per paragraph 2. above.

(1) Position a 2-inch (5 centimeter) wide strip of masking tape over new vane plate and mark an arc on the tape at the trailing edge of the vane using a ballpoint pen.
(2) Perform "Functional Test of Stall Warning System" per 27-31-00 Maintenance Practices, paragraph 1.A. or 1.D., adding marks to tape per figure 201 at actuation points.
(3) Repeat functional test to check marks on tape.
(4) Scribe marks except arc through masking tape.

B. Check Vane Plate Markings *Aircraft 35-279 and Subsequent and 36-045 and Subsequent and prior Aircraft equipped with Stall Fence (AAK 79-10 or AMK 83-5) or AAK 83-2, "Installation of FC-530 Autopilot*

NOTE: If stall warning pusher and shaker actuation is not within the tolerance specified in 27-31-00 Maintenance Practices, paragraph 2.A or 2.D the vane plate marks can be verified for accuracy using the following procedure. Installation and marking of a new vane plate may be necessary.

On *Aircraft 35-279 thru 35-505 and 36-045 thru 36-053 and prior aircraft modified per AAK 79-10 or AMK 83-5, "Stall Fences,"* mylar templates, P/N 5471102-3 (LH) and P/N 5471102-4 (RH), are provided to verify mark location and to assist in remarking the plate while the plate is installed on the airplane. When correctly marked, the vane plate provides a reference for easily testing the stall warning system for calibration accuracy on subsequent maintenance inspection cycles.

On *Aircraft 35-506 and Subsequent and 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot,"* mylar templates 5471102-7 (LH) and 5471102-8 (RH), are provided to verify mark location and to assist in remarking the plate while the plate is installed on the airplane. When correctly marked, the vane plate provides a reference for easy testing the stall warning system for calibration accuracy on subsequent maintenance inspection cycles.

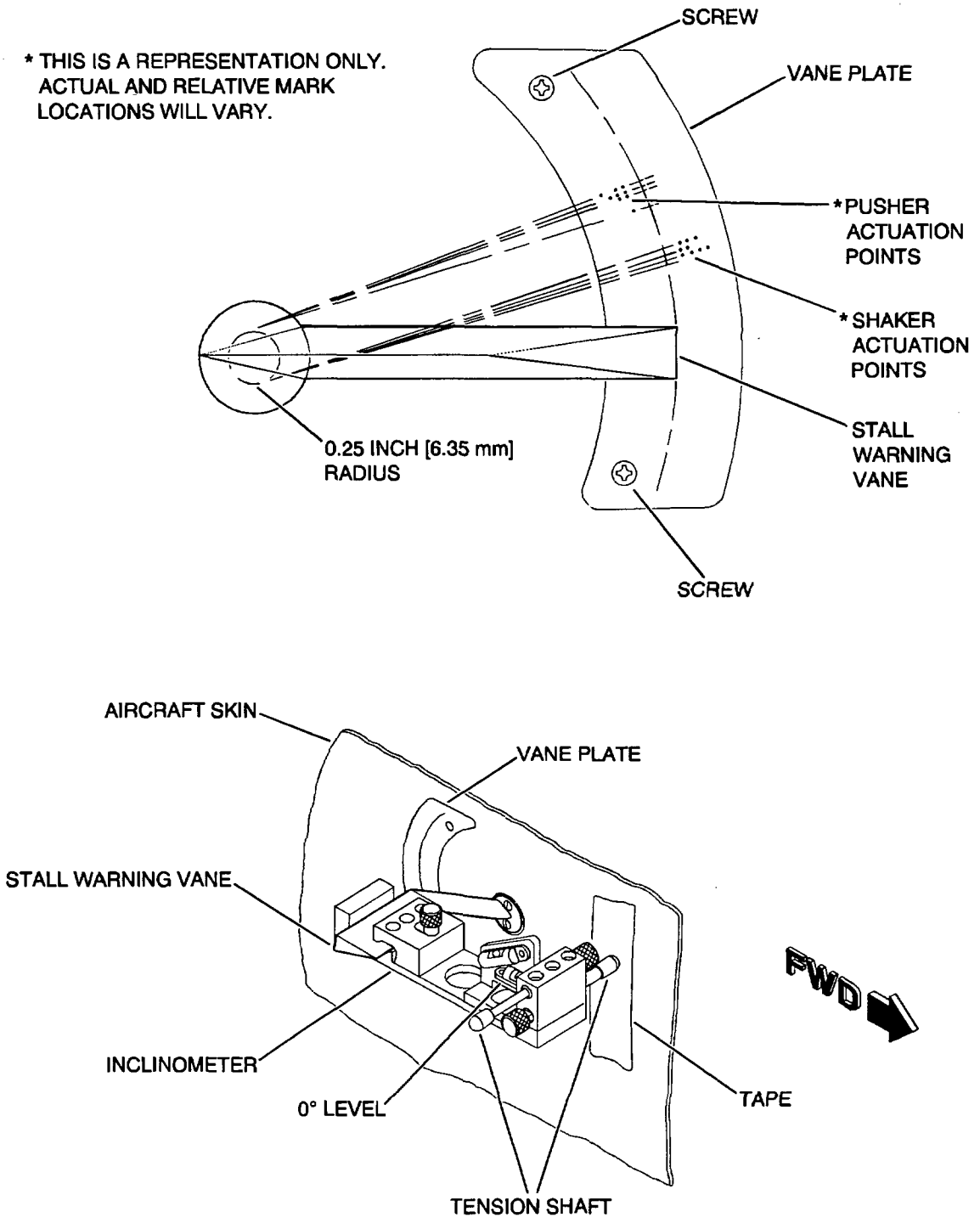
This section describes the use of mylar templates to aid in marking stall plates.

(1) Using jacks, level the airplane. (Refer to Chapter 8.)
(2) If vane plate is not marked by an arc from top to bottom under trailing edge of vane, use a scribe taped to trailing edge of vane to carefully scribe an arc for reference.
(3) Place a piece of tape forward of angle-of-attack vane so that rubber-tipped tension shaft of inclinometer will not mar painted surface. Install vane positioning inclinometer, on vane and position vane to zero degrees. Secure tool finger-tight only.
(4) Place vane plate marking template over vane plate and align it with arc on the vane plate. Next, position mylar template along arc until template reference line is aligned with top edge of vane. Secure mylar template in this position.

(5) Remove vane position inclinometer for better access to mylar.
(6) Compare existing marks on vane plate with the corresponding marks on the template.
(7) If mark positions are not in agreement, remove old vane plate and install a new, blank vane plate. (Refer to paragraph 2, above.)
(8) Position the mylar template as described in steps (1) through (4) above.
(9) Before final scribing of marks, verify that shaker and pusher actuation occurs within prescribed tolerance of template marks.

NOTE: If all test points are uniformly in error, it is likely that the template is mislocated. If the error varies, it is likely that the computer is misadjusted. If computer misadjustment is suspected, perform the calibration per 27-31-00 Maintenance Practices, paragraph 2.H. prior to marking the vane.

(10) When correct template location and computer calibration are ensured, carefully scribe reference marks through template onto plate.
(11) Remove mylar template and scribe appropriate lines and symbols through reference marks on step (9).
(12) Perform the functional test per 27-31-00, paragraph 2.D., steps (13) thru (25); ensure that actuation of the shaker and pusher occurs at the correct position.



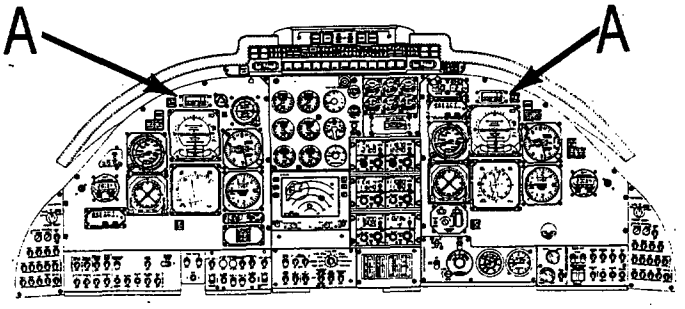
Stall Warning Vane Plate Installation and Marking
Figure 201

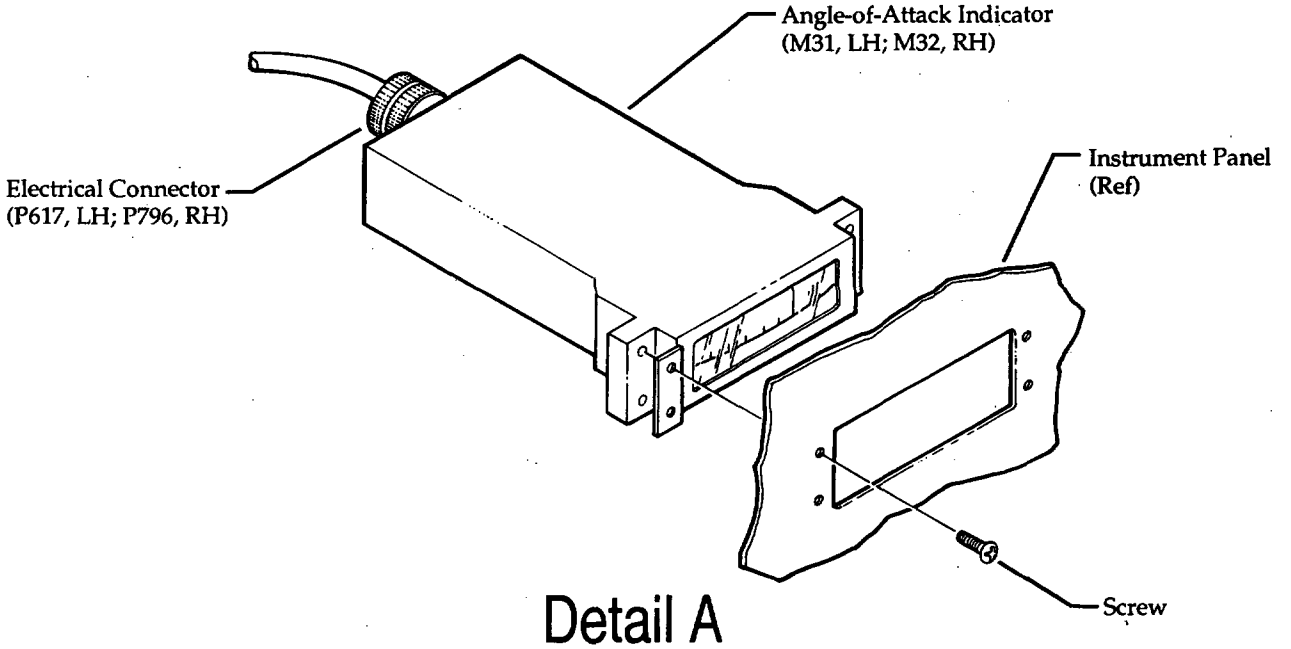
ANGLE-OF-ATTACK INDICATOR - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Angle-of-Attack Indicator (M31, LH; M32, RH) (See Figure 201.)
   (1) Remove electrical power from aircraft.
   (2) Lower applicable instrument panel.
   (3) Disconnect electrical connector (P617, LH; P796, RH) from angle-of-attack indicator.
   (4) Remove attaching parts securing indicator to instrument panel and remove from aircraft.

B. Install Angle-of-Attack Indicator (M31, LH; M32, RH) (See Figure 201.)
   (1) Position angle-of-attack indicator on instrument panel and secure with attaching parts.
   (2) Connect electrical connector (P617, LH; P796, RH) to angle-of-attack indicator.
   (3) Raise and secure instrument panel.
   (4) Restore electrical power to aircraft.
   (5) Perform functional test of stall warning system. (Refer to Adjustment/Test, 27-31-00.)





16-99A

Angle-of-Attack Indicator Installation
Figure 201

HORIZONTAL STABILIZER - DESCRIPTION AND OPERATION

1. DESCRIPTION

A. The horizontal stabilizer control system provides pitch trim control in primary and secondary modes.

B. On Aircraft 35-002 thru 35-505, 36-002 thru 36-053 not modified per AAK 83-2, "Installation of FC-530 Autopilot," the horizontal stabilizer control system consists of the horizontal stabilizer, an actuator, a pitch trim position indicator, a Pitch Trim Select Switch, a Secondary Pitch Trim Switch, a Pilot's and Copilot's Control Wheel Trim Switch, a Pilot's and Copilot's Pitch Trim Arming Pushbutton, and utilizes a number of relays in the trim switch panel.

C. On Aircraft 35-506 and Subsequent, 36-054 and Subsequent, and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot," the horizontal stabilizer control system consists of the horizontal stabilizer, a two-speed actuator, a pitch trim position indicator, a pitch trim speed controller, a Pitch Trim Select Switch, a Secondary Pitch Trim Switch, a Pilot's and Copilot's Pitch Trim Switch, a Pilot's and Copilot's Control Wheel Pitch Trim Switch, a Pilot's and Copilot's Control Wheel Trim Arming Pushbutton, relays in the trim switch panel, overload sensor on tailcone panel, a PITCH TRIM annunciator, and the Trim-in-Motion System (refer to 27-41-00 for maintenance of the Trim-in-Motion System).

D. Component Description

(1) The movable horizontal stabilizer serves as the pitch trim controller. The horizontal stabilizer is attached to the vertical stabilizer at two points. The aft attachment is the pivot point and consists of a large hinge pin inserted through heavy bearings in the horizontal and vertical stabilizers. The forward attach point is a heavy-duty electrical actuator.

(2) The screwjack-type actuator adjusts the horizonal stabilizer to provide aircraft longitudinal trim. The actuator is installed on vertical stabilizer spar No. 2 at WL 96.

(3) The pitch trim position indicator is installed on the pedestal.

(4) On Aircraft 35-506 and Subsequent, 36-054 and Subsequent, the pitch trim speed controller is located in the vertical stabilizer forward of spar No. 2 at WL 85.

(5) The Pitch Trim Select Switch, located on the trim switch panel, is a three-position toggle switch used to select primary or secondary modes of operation. The Pitch Trim Select Switch is marked PITCH TRIM/PRI/SEC/OFF.

(6) The Secondary Pitch Trim Switch, located on the trim switch panel on the pedestal, is a three-position toggle switch used to manually change pitch when the Pitch Trim Select Switch is set to SEC. The Secondary Pitch Trim Switch is marked NOSE UP/OFF/NOSE DN.

(7) The Pilot's and Copilot's Control Wheel Trim Switches are mounted on the control wheels. The Control Wheel Trim Switches are marked NOSE DN, NOSE UP, LWD and RWD.

(8) The Pilot's and Copilot's Control Wheel Trim Arming Pushbuttons are mounted on the control wheels in the center of the Control Wheel Trim Switches.

(9) Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot," the overload sensor (horizontal stabilizer actuator) is located on the overload sensor panel on the LH side of the tailcone aft of frame 26.

(10) Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot," the PITCH TRIM annunciator is located on the glareshield adjacent to the left fire pull handle.

2. OPERATION

NOTE: Primary and secondary horizontal stabilizer trim operations are described here. Horizontal stabilizer trim using the autopilot is described in Chapter 22.

A. On Aircraft 35-002 thru 35-505, 36-002 thru 36-053, the pitch trim operation is as follows:

(1) With the Pitch Trim Select Switch set to PRI, a ground circuit is completed through the Pitch Trim Select Switch to the de-energized contacts of the trim control relay. Twenty-eight vdc is applied from the pitch trim circuit breakers to the Pilot's Control Wheel Trim Switch and Pilot's Control Wheel Trim Arming Pushbutton. When the Pilot's Control Wheel Trim Arming Pushbutton is depressed and the Pilot's Control Wheel Switch is set to either NOSE UP or NOSE DN, 28 vdc is applied to the primary pitch trim motor (M1), the pilot's authority relay, and the trim control relay. The pilot's authority relay energizes and removes power from the Copilot's Control Wheel Trim Switch and the Copilot's Control Wheel Trim Arming Pushbutton, while the trim control relay energizes, completing the ground circuit to the primary pitch trim motor (M1). Operation of the copilot's primary pitch trim system is identical to the pilot's primary pitch trim system, except 28 vdc is applied through the de-energized contacts of the pilot's authority relay to the Copilot's Control Wheel Trim Switch and Copilot's Control Wheel Trim Arming Pushbutton. During primary trimming operation the autopilot will be disengaged.

(2) With the Pitch Trim Select Switch set to SEC, the ground source for the primary pitch trim motor is removed, which makes the pitch trim function of the Control Wheel Trim Switches and Control Wheel Trim Arming Pushbuttons inoperative. During secondary pitch trim operation, a ground circuit is completed through the Pitch Trim Select Switch to the secondary pitch trim motor (M2). Twenty-eight vdc (either NOSE UP or NOSE DN) is applied through the Pitch Trim Select Switch to the secondary pitch trim motor (M2). The autopilot will disengage when 28 vdc (either NOSE UP or NOSE DN) is applied through the Pitch Trim Select Switch to the secondary pitch trim motor (M2).

3. The pitch trim actuator incorporates a switch which lights the T.O. TRIM (amber) annunciator on the glareshield. The light alerts the crew that the horizontal stabilizer is not properly trimmed for take-off. After the aircraft is airborne, the squat switch opens the circuit and the caution light becomes inoperative.

B. Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot," the pitch trim operation is as follows:

(1) Primary Operation

(a) With the Pitch Trim Select Switch set to PRI, a standby ground circuit is completed through the Pitch Trim Select Switch to the de-energized contacts of the trim control relay. Twenty-eight vdc is applied from the PITCH circuit breaker to the Pilot's Control Wheel Trim Switch and Trim Arming Pushbutton. When the Pilot's Control Wheel Trim Switch and Trim Arming Pushbutton is depressed and set to either NOSE UP or NOSE DN, 28 vdc is applied to the pilot's authority relay, the trim control relay, and the pitch trim actuator primary (M1) motor through the trim speed controller. The pilot's authority relay energizes to remove power from the Copilot's Control Wheel Trim Switch and Trim Arming Pushbutton, while the trim control relay energizes to complete a circuit to the pitch trim actuator primary motor through the pitch trim speed controller and then to ground.

(b) The low speed of the primary motor is achieved by reducing current through the trim speed controller when the flaps are up. Should a failure occur which would allow full-speed motor operation with the flaps up, a self-monitoring circuit will illuminate the Pitch Trim warning light on the glareshield.

(c) At takeoff, with the squat switches in ground mode, if the horizontal stabilizer pitch angle is beyond prescribed take-off up or down limits, the applicable up or down limit switch in the stabilizer actuator linear potentiometer will be closed, making a circuit to ground through the left squat switch to the T.O. TRIM annunciator.

(d) With the flaps up and the Pitch Trim Select Switch set to PRI, 65 to 69 seconds are required for the actuator to move from stop to stop. With the flaps down and the Pitch Trim Select Switch set to PRI, 14 to 20 seconds are required for the actuator to move from stop to stop.

(e) The actuator incorporates a linear potentiometer which follows actuator motion. The potentiometer supplies a signal to the pitch trim position indicator and modulates a circuit to the trim-in-motion detector. (Refer to 27-41-00.)

(f) During normal operation, the speed controller and pitch trim actuator primary motor are energized through the closed contacts of the pitch trim actuator overload sensor relay and the PITCH circuit breaker. If an overload condition exists, the thermal breaker contained within the sensor will position the switch contacts to remove power from the relay and ground the PITCH

circuit breaker. The overload sensor will reset when the thermal breaker has cooled. However, the 28 vdc required to energize the relay will not be present due to the open PITCH circuit breaker. After the malfunction has been corrected and the PITCH circuit breaker reset, the relay will energize.

(2) Secondary Operation

(a) With the Pitch Trim Switch set to SEC, the Pilot's and Copilot's Control Wheel Trim Switches and Trim Arming Pushbuttons are rendered inoperative. Stabilizer trim is accomplished by setting the Secondary Pitch Trim Switch to either NOSE UP or NOSE DN. This directs 28 vdc to the pitch trim actuator secondary motor (M2) which operates at a fixed low speed. During secondary operation, a ground circuit is completed through the contacts of the Pitch Trim Select Switch to the pitch trim actuator secondary motor. With the flaps down and the Pitch Trim Select Switch set to SEC, 68 seconds are required for the actuator to move from stop to stop.

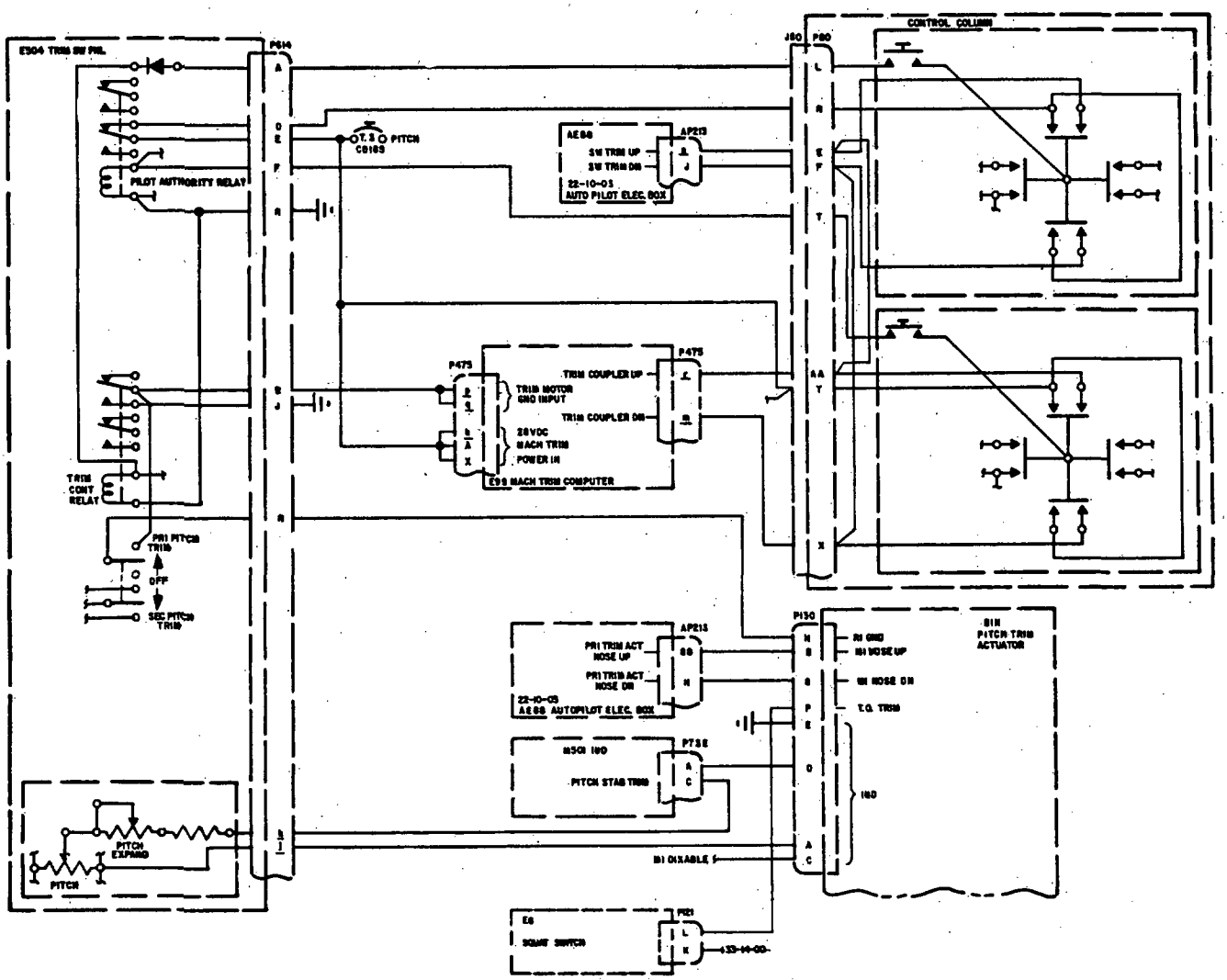
(b) The horizontal stabilizer control circuitry is biased through the right flap switch which closes as the flaps are extended through 3°, shifting the stabilizer actuator operational mode from low to high speed. Actuator speed monitoring is accomplished through two magnetic pickups in the primary motor.

(3) Primary and Secondary Operation

(a) Operation of the horizontal stabilizer control system in either primary or secondary mode will disengage the autopilot.

(b) Depressing the Control Wheel Master Switch energizes two trim disable relays in the pitch trim actuator, rendering both motors inoperative.

(c) Refer to Chapter 22 for autopilot pitch trim information.

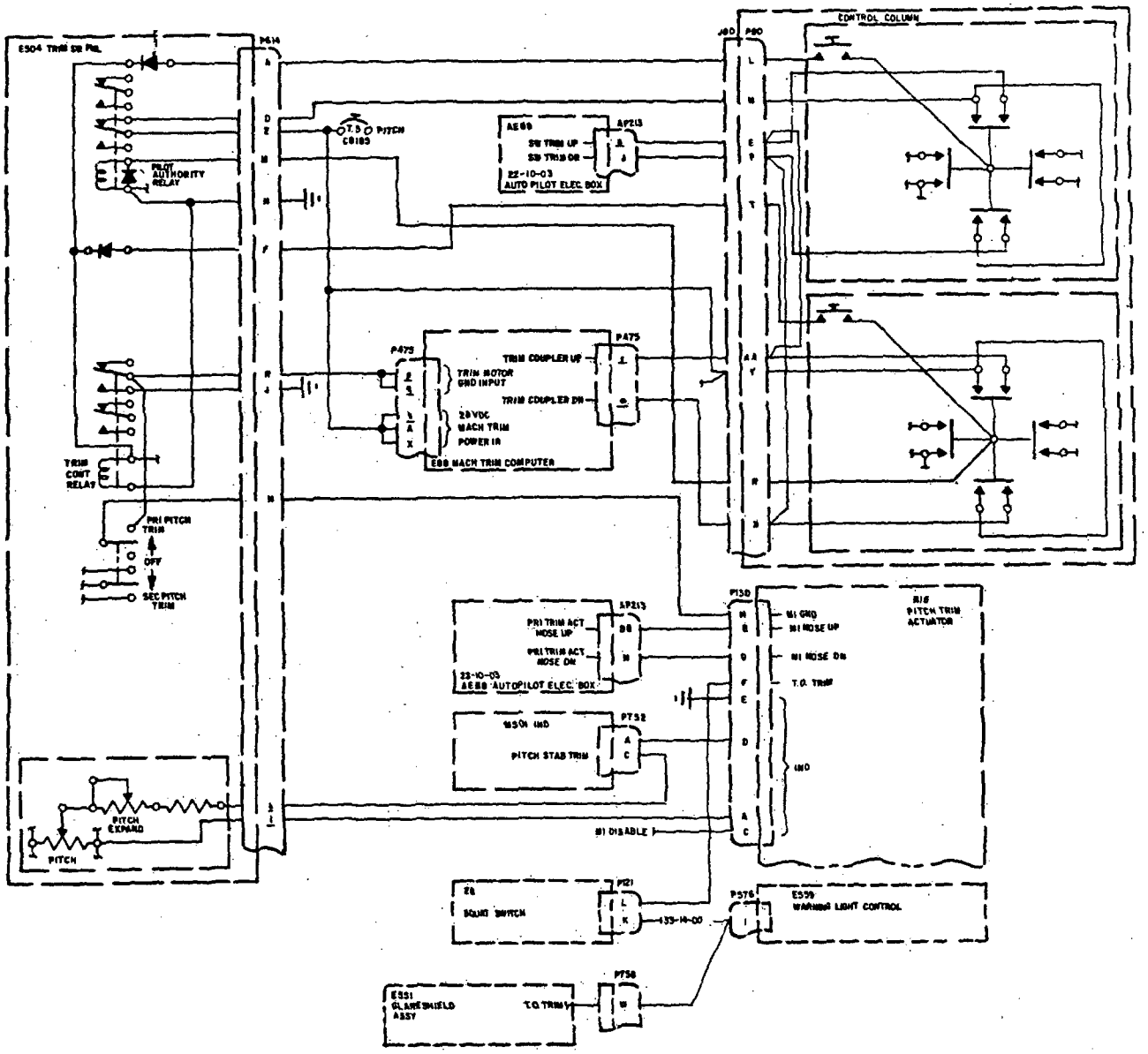


PRIMARY PITCH TRIM

Pitch Trim System Electrical Control Schematic
Figure 1 (Sheet 1 of 7)

EFFECTIVITY: 35-002 thru -048; 36-002 thru -016, when equipped with double width pedestal and not modified per AAK 83-2, "Installation of FC-530 Autopilot"

MM-99
Disk 539

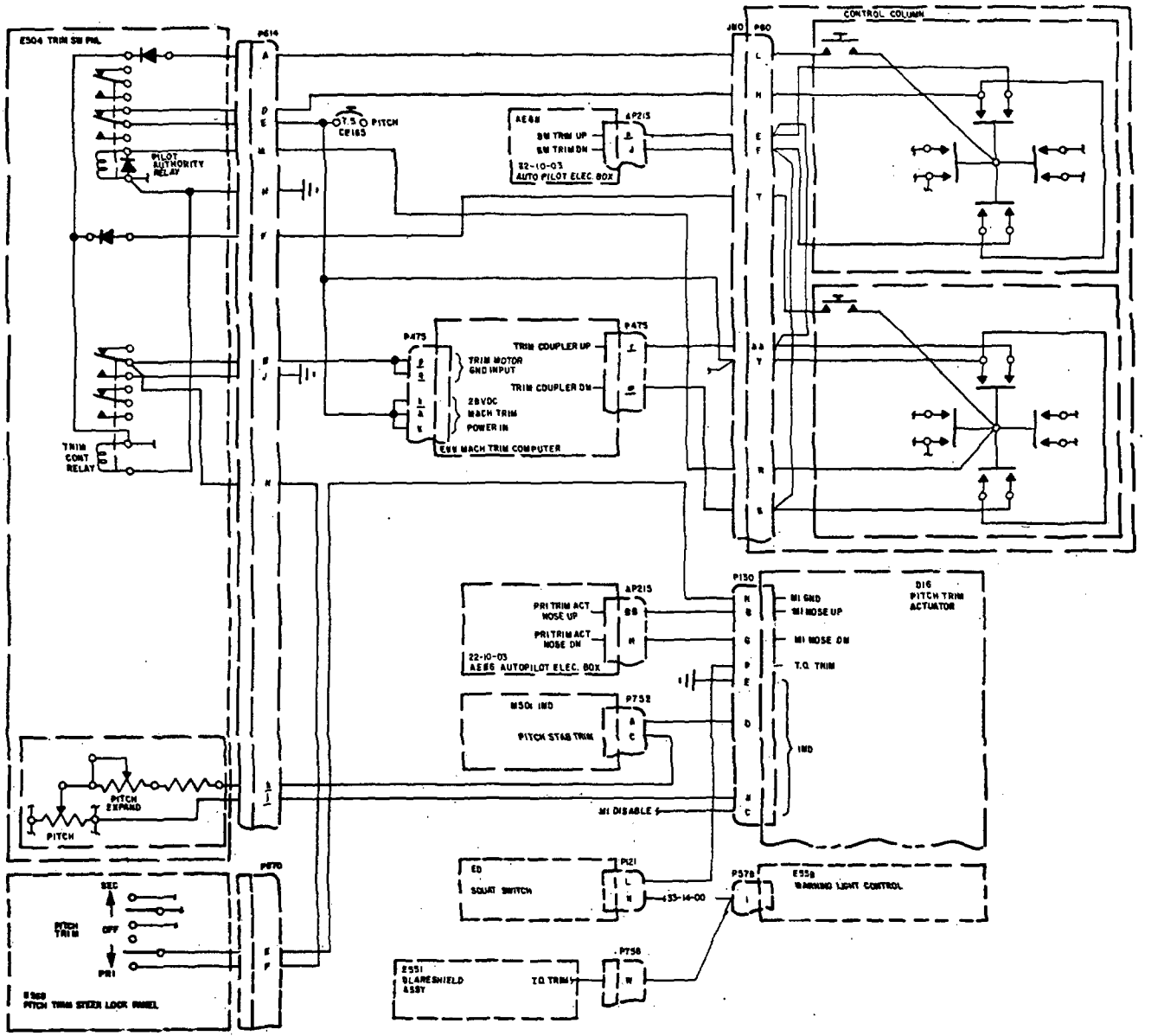


PRIMARY PITCH TRIM

Pitch Trim System Electrical Control Schematic
Figure 1 (Sheet 2 of 7)

EFFECTIVITY: 35-049 thru -070; 36-017 and -018 when equipped with double width pedestal; 35-049 thru 35-505, 36-017 thru 36-053 when equipped with single width pedestal and not modified per AAK 83-2, "Installation of FC-530 Autopilot"

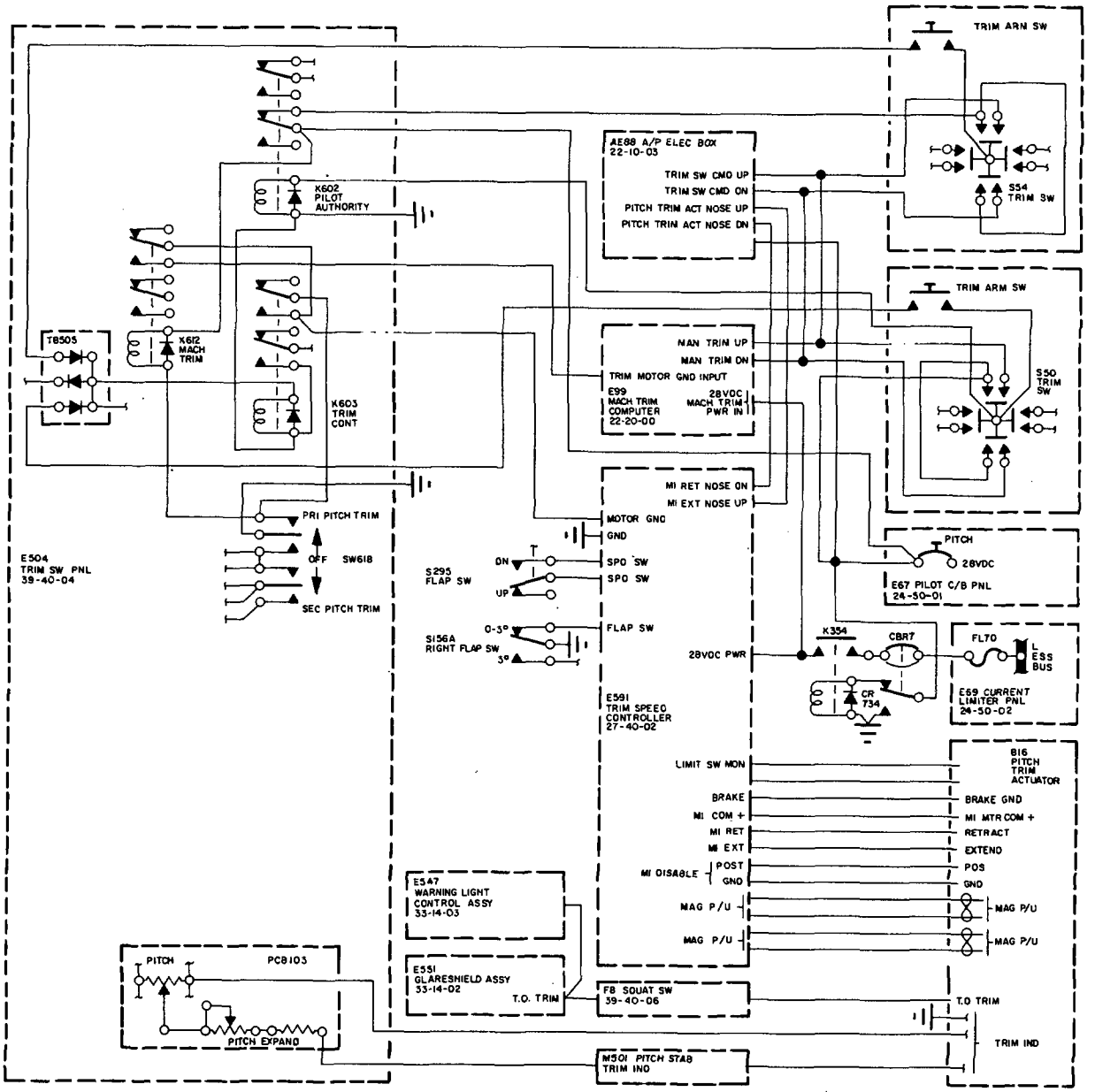
MM-99
Disk 539



PRIMARY PITCH TRIM

Pitch Trim System Electrical Control Schematic
Figure 1 (Sheet 3 of 7)

EFFECTIVITY: 35-071 thru 35-505, 36-019 thru 36-053 when equipped with double width pedestal and not modified per AAK 83-2, "Installation of FC-530 Autopilot"

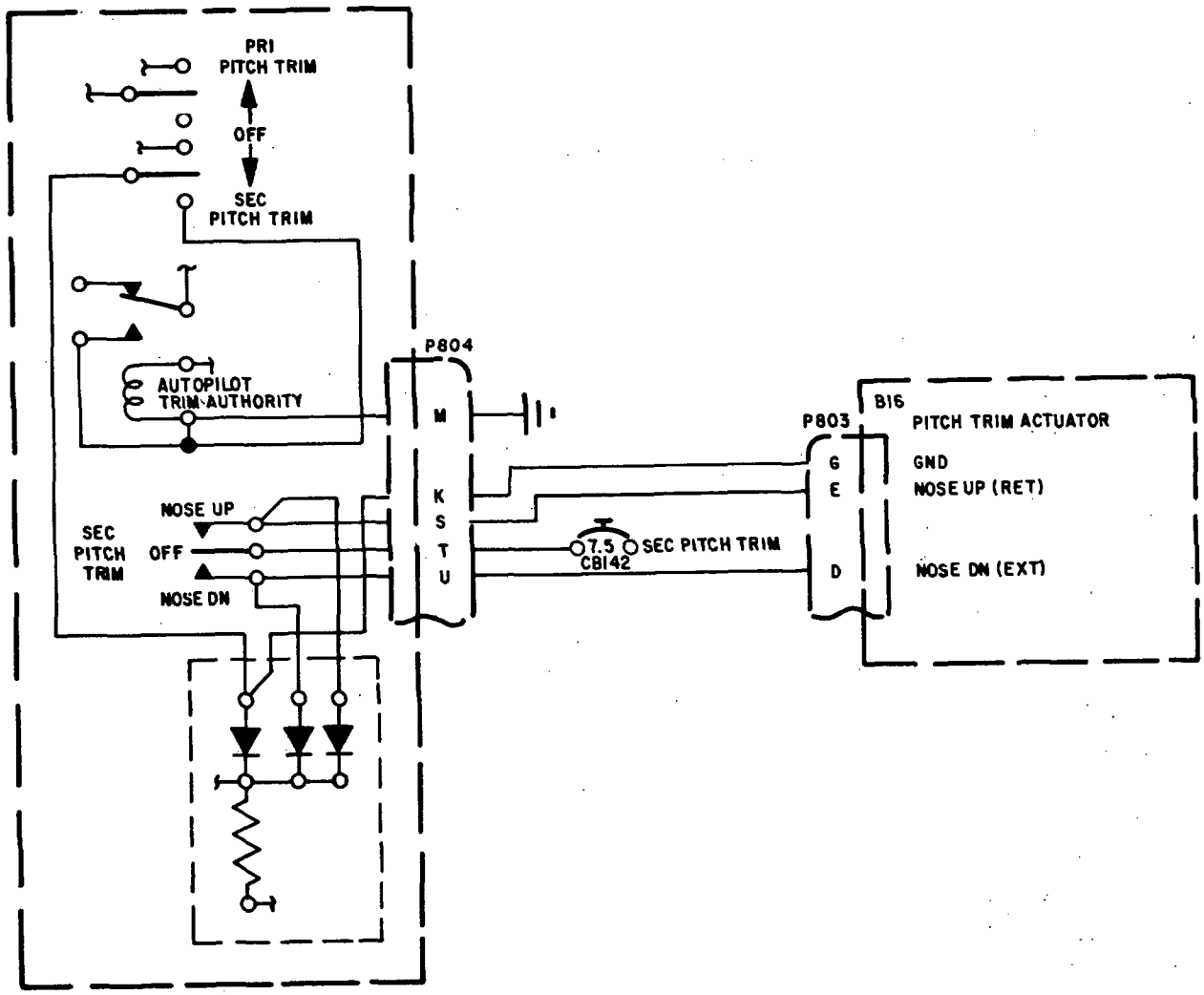


PRIMARY PITCH TRIM

Pitch Trim System Electrical Control Schematic
Figure 1 (Sheet 4 of 7)

EFFECTIVITY: 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2, "Installation of FC-530 Autopilot"

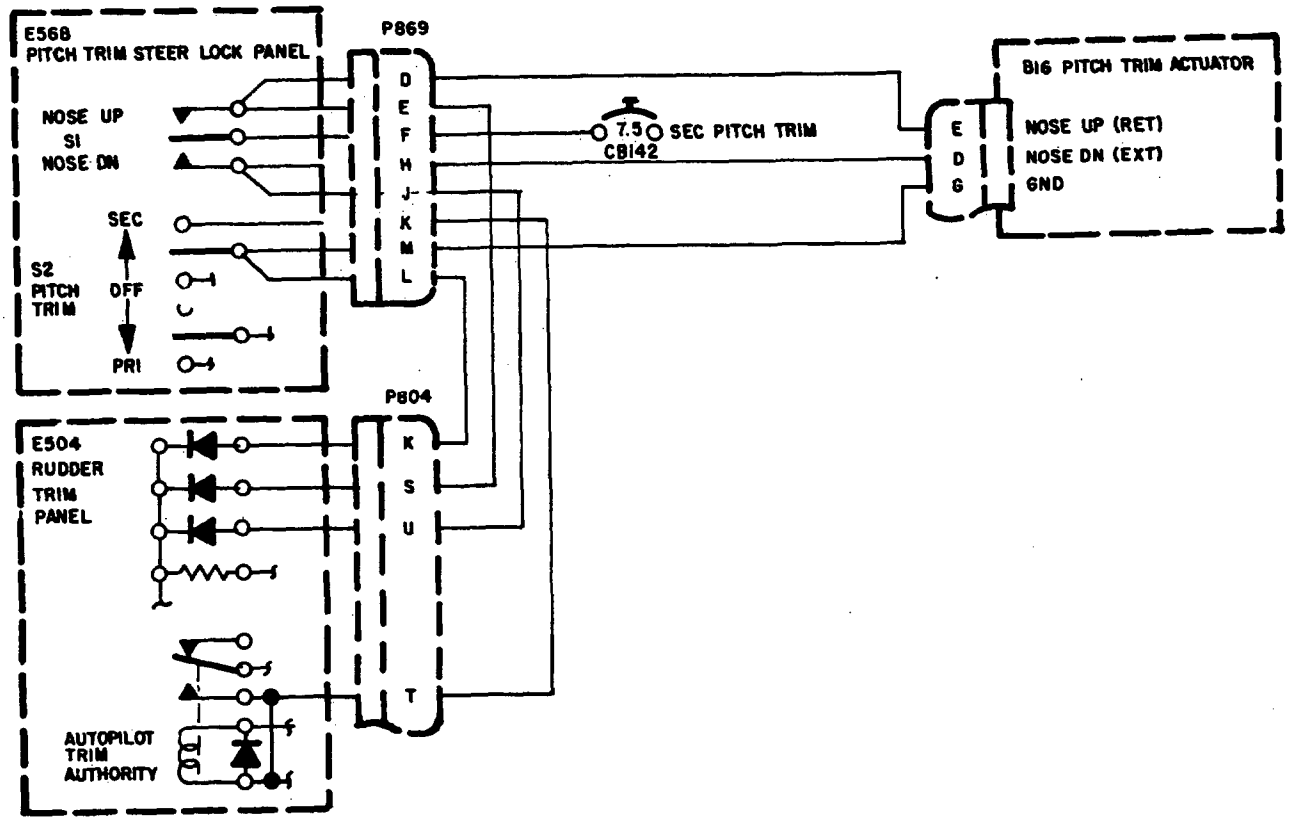
MM-99
Disk 539



SECONDARY PITCH TRIM

Pitch Trim System Electrical Control Schematic
Figure 1 (Sheet 5 of 7)

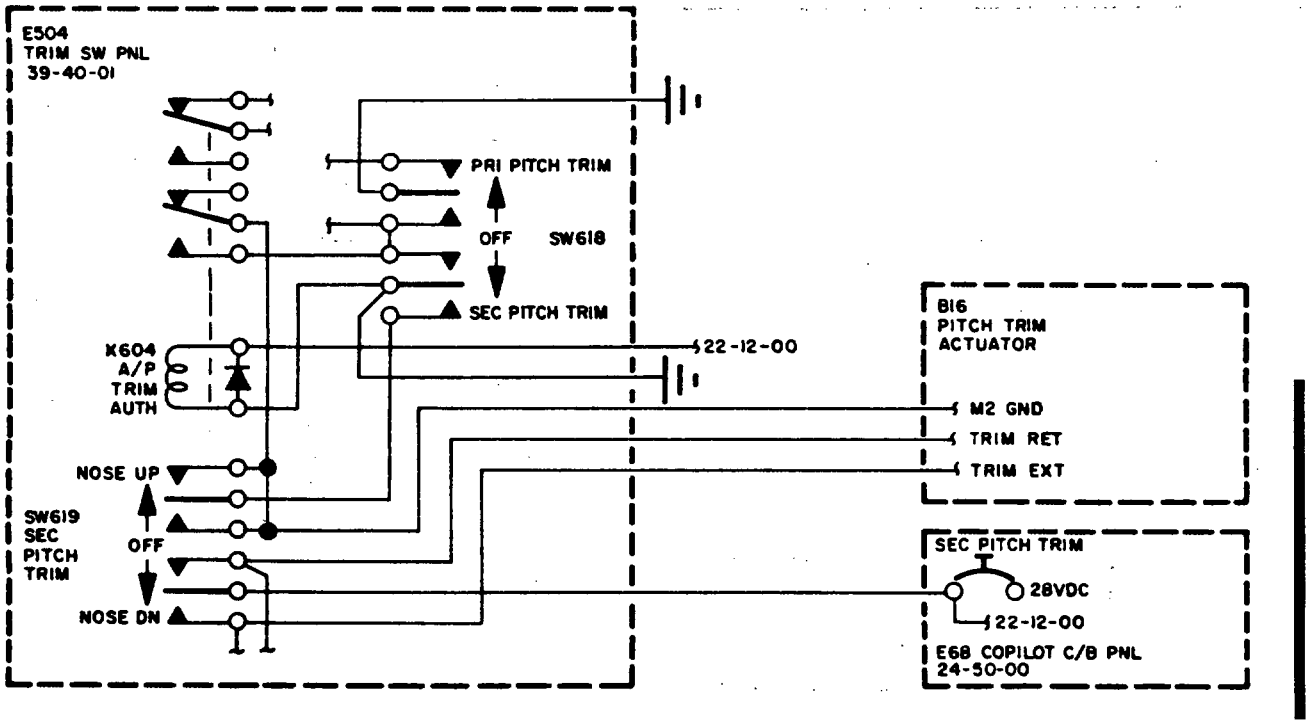
EFFECTIVITY: 35-002 THRU -070; 36-002 THRU -018 when equipped with double width pedestal; 35-002 thru 35-505, 36-002 thru 36-053 when equipped with single width pedestal and not modified per AAK 83-2, "Installation of FC-530 Autopilot"



SECONDARY PITCH TRIM

Pitch Trim System Electrical Control Schematic
Figure 1 (Sheet 6 of 7)

EFFECTIVITY: 35-017 thru 35-505; 36-019 thru 36-053 when equipped with double width pedestal and not modified per AAK 83-2, "Installation of FC-530 Autopilot"



(SECONDARY PITCH TRIM)

Pitch Trim System Electrical Control Schematic
Figure 1 (Sheet 7 of 7)

HORIZONTAL STABILIZER CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	Measure angles.
Digital Protractor (DP-45) (Alternate)	02475-01	Lucas & Schaebitz Pennsauken, NJ	Measure angles.
Contour Board	T/N 15785	Learjet Inc. Wichita, KS	Rigging.
Rigging Pin	3170037-16	Learjet Inc. Wichita, KS	Rigging.

2. Adjustment/Test

NOTE: During rigging procedures the stabilizer position indicator is to be used for reference only. The upper screw is located 1° 40' down from neutral and the lower screw is located 8° 7' down from neutral.

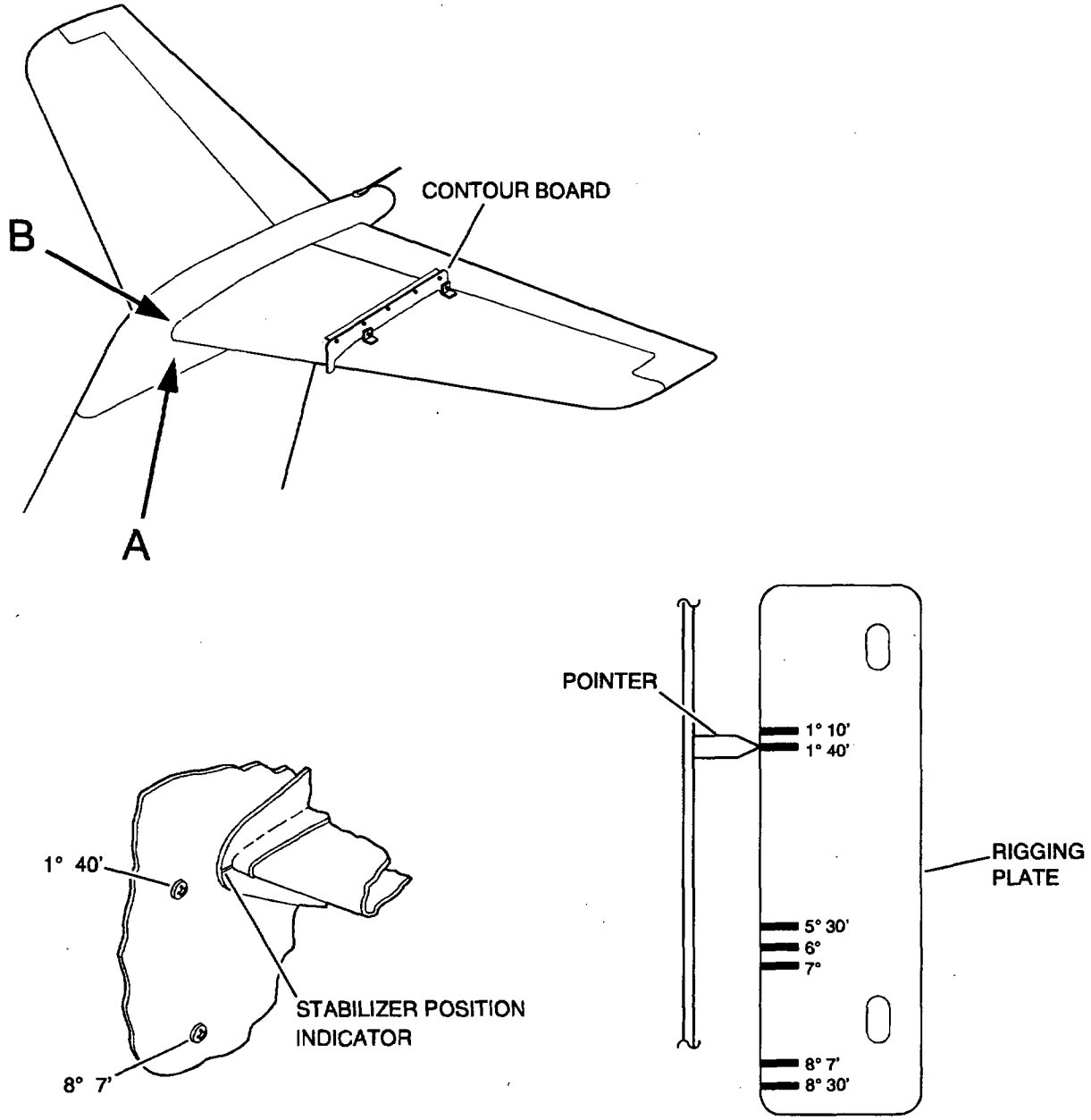
A. Rig Horizontal Stabilizer
(1) Jack aircraft. (Refer to Chapter 7.)
(2) Level aircraft. (Refer to Chapter 8.)
(3) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.
(4) Remove forward vertical stabilizer tip access cover from vertical stabilizer.

NOTE: *On Aircraft not modified per SB 35/36-27-21*, a master rigging plate and position indicator, located on vertical stabilizer inner structure, is provided as an aid to check stabilizer position during maintenance. The rigging plate is marked to indicate allowable stabilizer travel tolerance.

If rigging plate has been removed or pointer has been bent or broken and repaired, proceed with step (5). If rigging plate and/or pointer has not been damaged, proceed to step (12).

On Aircraft 35-001 thru 35-676 and 36-001 thru 36-063 modified per SB 35/36-27-21, if rigging bracket is installed as an aid to check stabilizer position during maintenance, proceed to step (12).

(5) Drill out rivets which attach the rigging plate to the aircraft and remove rigging plate.
(6) Ensure that pointer is free to move through entire travel of horizontal stabilizer without interference or binding.
(7) Remove two countersunk screws at LBL 34.80 and install contour board mounting brackets and contour board.
(8) Place vernier clinometer on the level plate of the contour board.
(9) Move stabilizer to 6° (stabilizer leading edge down from neutral) and scribe a mark extending the path that pointer would make across rigging plate.



AIRCRAFT 35-002 THRU 35-066 AND 36-002 THRU 36-017 NOT EQUIPPED WITH AAK 76-4, "REDUCED APPROACH SPEED SYSTEM".

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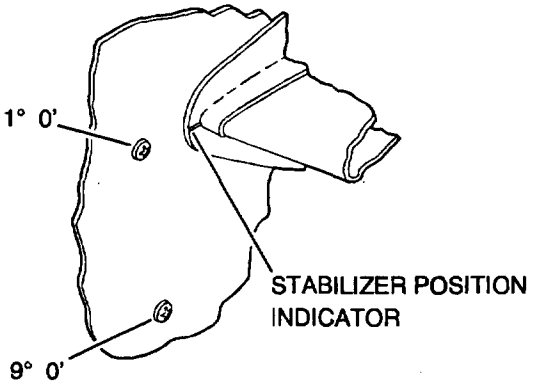
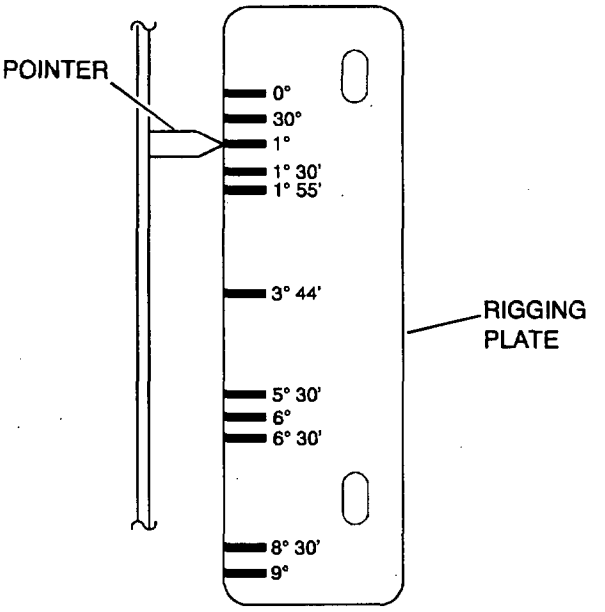
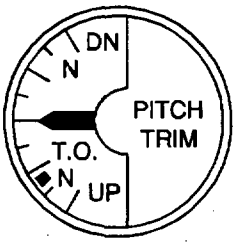


Detail A

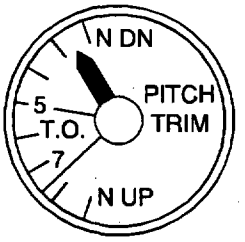


Detail B

AIRCRAFT 35-067 AND SUBSEQUENT, 36-018 AND SUBSEQUENT, AND EARLIER AIRCRAFT EQUIPPED WITH AAK 76-4, "REDUCED APPROACH SPEED SYSTEM".



(PITCH TRIM INDICATOR)



(PITCH TRIM INDICATOR)

AIRCRAFT 35-002 THRU 35-145 AND 36-002 THRU 36-035, NOT MODIFIED BY AAK 83-2, "INSTALLATION OF FC-530 AUTOPILOT" OR AAK 83-8, "INSTALLATION OF COMPONENTS TO ALLOW VARIABLE PITCH TRIM SETTING AT TAKEOFF".

AIRCRAFT 35-146 AND SUBSEQUENT, 36-036 AND SUBSEQUENT, AND PRIOR AIRCRAFT MODIFIED BY AAK-83-2, "INSTALLATION OF FC-530 AUTOPILOT" OR AAK 83-8, "INSTALLATION OF COMPONENTS TO ALLOW VARIABLE PITCH TRIM SETTING AT TAKEOFF".

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Horizontal Stabilizer Rigging
Figure 201 (Sheet 2 of 2)

(10) Move stabilizer to full up position and extend mark scribed in step (9) to the right of where rigging plate is to be mounted.

(11) Position rigging plate so that 6° line on rigging plate aligns (vertically) with line scribed in step (10). Ensure that mounting holes of rigging plate align (horizontally) with existing mounting holes in aircraft structure. If necessary, mark and drill mounting holes. Rivet rigging plate in place.

(12) Set Pitch Trim Selector Switch to NORM and operate primary pitch trim system using Pilot's or Copilot's Trim and Trim Arming Switch to NOSE UP or NOSE DN. Observe horizontal stabilizer movement for proper operation and correct travel.

NOTE: The allowable tolerance for stabilizer travel is: On *Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not modified per AAK 76-4, "Reduced Approach Speed System"*, 1° 10' to 1° 40' stabilizer leading edge down from neutral and 8° 7' to 8° 30' stabilizer leading edge down from neutral. On *Aircraft 35-067 and Subsequent, 36-018 and Subsequent, and prior aircraft modified per AAK 76-4, "Reduced Approach Speed System"*, 1° 30' to 1° 55' stabilizer leading edge down from neutral and 8° 30' to 9° 00' stabilizer leading edge down from neutral.

(13) Adjust actuator rod end as necessary. One complete turn of rod end changes actuator travel 10 minutes.

CAUTION: SECURE BOTH ENDS OF STABILIZER ACTUATOR BEFORE OPERATION. IF UPPER ROD END IS NOT SECURED, POTENTIOMETER ROD MAY BE DAMAGED BY TWISTING MOTION (TORQUE) OF ACTUATOR SHAFT.

(14) Attach actuator rod end to stabilizer and operate Pitch Trim Switch on pedestal. Check stabilizer for proper operation and travel.

(15) Ensure that pitch trim indicator is properly calibrated. (Refer to 27-40-01.)

(16) On *Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not modified per AAK 76-4, "Reduced Approach Speed System"*, actuate horizontal stabilizer to 6.5° leading edge down. The pitch trim indicator will point to the green band. The T.O. TRIM annunciator on glareshield shall not be illuminated.

NOTE: A clinometer will have to be used to measure the 6.5° down for horizontal stabilizer leading edge.

(17) On *Aircraft 35-067 and Subsequent, 36-018 and Subsequent, and prior aircraft modified per AAK 76-4, "Reduced Approach Speed System"*, actuate horizontal stabilizer to 7.4° leading edge down. On *Aircraft 35-067 thru 35-145 and 36-018 thru 36-035, not modified by AAK 83-2, "Installation of FC-530 Autopilot", or AAK 83-8, "Installation of Components to Allow Variable Pitch Trim Setting at Takeoff"*, the pitch trim indicator will point to the green band. On *Aircraft 35-146 and Subsequent and 36-036 and Subsequent, and prior aircraft modified by AAK 83-2, "Installation of FC-530 Autopilot", or AAK 83-8, "Installation of Components to Allow Variable Pitch Trim Setting at Takeoff"*, the trim indicator will point to 7.4°. The T.O. TRIM annunciator on glareshield shall not illuminate.

NOTE: A clinometer will have to be used to measure the 7.4° down for horizontal stabilizer leading edge.

(18) Install forward vertical stabilizer tip-access cover and both horizontal stabilizer fairings.

3. Inspection/Check

NOTE: Perform takeoff trim operational check in accordance with the current inspection intervals as specified in Chapter 5.

A. Takeoff Trim Operational Check *(35-002 thru 35-066, 36-002 thru 36-017, not modified per AAK 76-4, "Reduced Approach Speed System", AAK 83-2 "Installation of FC-530 Autopilot", or AAK 83-8, "Installation of Components to Allow Variable Pitch Trim Setting at Takeoff")*
   (1) Place aircraft on jacks and level. (Refer to Chapter 8.)
   (2) Block squat switches to ground mode.
   (3) Attach a contour board and clinometer to horizontal stabilizer. (Refer to Adjustment/Test, this section.)
   (4) Using either pilot's or copilot's Trim and Trim Arming Switch, position horizontal stabilizer to full up position and check that T.O. TRIM annunciator is illuminated.
   (5) Using either pilot's or copilot's Trim and Trim Arming Switch, position horizontal stabilizer down until T.O. TRIM annunciator is extinguished. Clinometer shall show horizontal stabilizer to be -5° 30' to -6° leading edge down.
   (6) Continue to position horizontal stabilizer down until T.O. TRIM annunciator is illuminated. Clinometer shall show horizontal stabilizer to be -7° to -7° 30' leading edge down.
   (7) If horizontal stabilizer position is not within the tolerances given in steps (4) and (5), the stabilizer actuator must be removed and returned to an authorized repair facility for adjustment. (Refer to applicable Horizontal Stabilizer Actuator Repair Manual for adjustment procedures.)

B. Takeoff Trim Operational Check *(35-067 thru 35-145, 36-018 thru 36-035, and prior aircraft modified per AAK 76-4 "Reduced Approach Speed System" but not modified per AAK 83-2 "Installation of FC-530 Autopilot" or AAK 83-8 "Installation of Components to Allow Variable Pitch Trim Setting at Takeoff")*
   (1) Place aircraft on jacks and level. (Refer to Chapter 8.)
   (2) Block squat switches to ground mode.
   (3) Attach a contour board and clinometer to horizontal stabilizer. (Refer to Adjustment/Test, this section.)
   (4) Using either pilot's or copilot's Trim and Trim Arming Switch, position horizontal stabilizer to full up position and check that T.O. TRIM annunciator is illuminated.
   (5) Using either pilot's or copilot's Trim and Trim Arming Switch, position horizontal stabilizer down until T.O. TRIM annunciator is extinguished. Clinometer shall show horizontal stabilizer to be -6° 36' to -7° 6' leading edge down.
   (6) Continue to position horizontal stabilizer down until T.O. TRIM annunciator is illuminated. Clinometer shall show horizontal stabilizer to be -7° 36' to -8° 6' leading edge down.
   (7) If horizontal stabilizer position is not within the tolerances given in steps (4) and (5), the stabilizer actuator must be removed and returned to an authorized repair facility for adjustment. Refer to applicable Horizontal Stabilizer Actuator Repair Manual for adjustment procedures.

C. Takeoff Trim Operational Check *(35-146 and Subsequent, 36-036 and Subsequent, and prior aircraft modified per AAK 83-2 "Installation of FC-530 Autopilot" or AAK 83-8 "Installation of Components to Allow Variable Pitch Trim Setting at Takeoff")*
   (1) Place aircraft on jacks and level. (Refer to Chapter 8.)
   (2) Block squat switches to ground mode.
   (3) Attach a contour board and clinometer to horizontal stabilizer. (Refer to Adjustment/Test, this section.)
   (4) Using either pilot's or copilot's Trim and Trim Arming Switch, position horizontal stabilizer to full up position and check that T.O. TRIM annunciator is illuminated.
   (5) Using either pilot's or copilot's Trim and Trim Arming Switch, position horizontal stabilizer down until T.O. TRIM annunciator is extinguished. Clinometer shall show horizontal stabilizer to be -4° 39' to -5° 9' leading edge down.
   (6) Continue to position horizontal stabilizer down until T.O. TRIM annunciator is illuminated. Clinometer shall show horizontal stabilizer to be -7° 24' to -7° 54' leading edge down.
   (7) If horizontal stabilizer position is not within the tolerances given in steps (4) and (5), the stabilizer actuator must be removed and returned to an authorized repair facility for adjustment. Refer to applicable Horizontal Stabilizer Actuator Repair Manual for adjustment procedures.

D. Horizontal Stabilizer Operational Check
   (1) Observe horizontal stabilizer through operational check for smooth, responsive movement.
   (2) Press down the Trim and Trim Arming Switch and set to NOSE DN position and hold.

(3) The horizontal stabilizer leading edge will move up from the neutral position.
(4) The pitch trim indicator on the center pedestal will indicate corresponding trim travel.
(5) Press down the Trim and Trim Arming Switch and set to NOSE UP position and hold.
(6) The horizontal stabilizer leading edge will move down from the neutral position.
(7) The pitch trim indicator will indicate corresponding trim travel.

E. Operational Check of Horizontal Stabilizer Control (Pitch Trim) System (*Aircraft modified per SB 35/36-27-21.*) (See Figure 202.)

NOTE: All readings in this operational check will be taken from the pitch trim position indicator.

(1) Set Battery Switches on.
(2) Set Pitch Trim Selector Switch, located on the Trim Switch Panel, to PRI.
(3) Check horizontal stabilizer travel as follows:
  (a) Using pilots control wheel Trim/Trim Arming Switch, trim horizontal stabilizer leading edge to full-up position.
  (b) Insert 0.25 inch [6.35 mm] rigging pin through holes in rigging brackets (located on horizontal stabilizer assembly) and through hole in rigging angle (located on fin fairing bulkhead assembly). This establishes the stabilizer leading edge up-travel position.

    NOTE: If necessary, adjust horizontal stabilizer actuator rod end to align holes on bracket and angle. One complete turn (360°) of the rod end changes the actuator travel 0.17° (10.2'). Do not adjust electrical limit switches or potentiometer shaft.

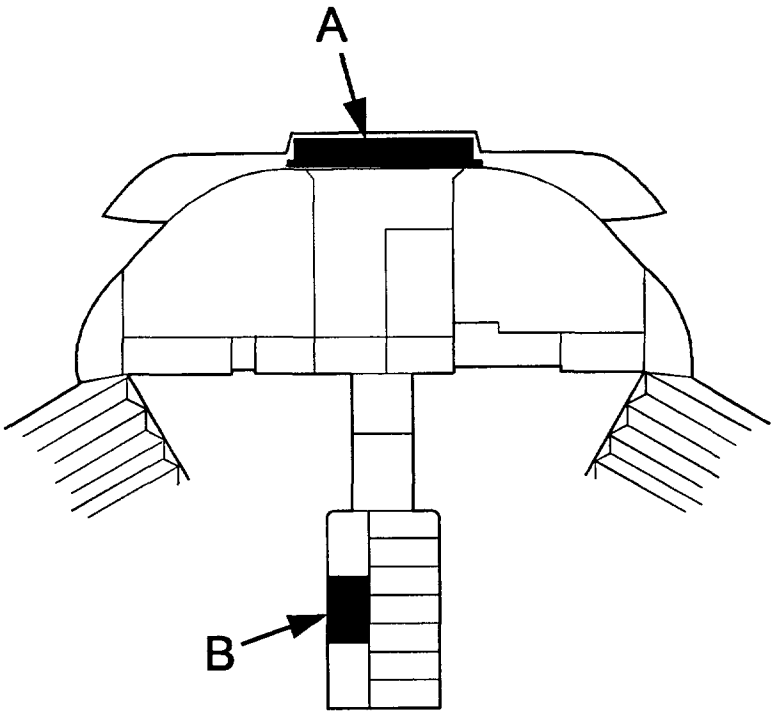
  (c) Pitch trim position indicator needle shall be at the appropriate position range mark (NDN, pinned position). (See Figure 202, Detail B).
  (d) Remove rigging pin.
  (e) If pitch position indicator needle does not indicate appropriate range mark (NDN, pinned position), adjust indicator. (Refer to 27-40-04, Adjustment/Test.)
  (f) Using pilots control wheel Trim/Trim Arming Switch, trim horizontal leading edge to full-down position.
  (g) Pitch trim indicator needle shall be at the appropriate position range mark (NUP). (See Figure 202, Detail B.)
  (h) If pitch position indicator does not indicate the appropriate position mark, (NUP), Adjust indicator.
  (i) Repeat steps E.(3)(a) thru (h) using copilots control wheel Trim/Trim Arming Switch.
(4) Check horizontal stabilizer T.O. Trim annunciator as follows:

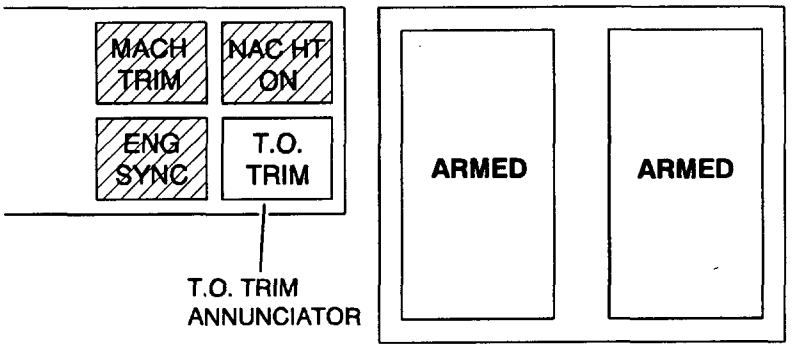
NOTE: Aircraft must be in ground mode (Squat switches closed) for operation of takeoff trim warning light.

If T.O. TRIM annunciator fails to illuminate or extinguish within these limits, adjust horizontal stabilizer position switch.

  (a) Using pilot's control wheel Trim/TrimArming Switch, trim horizontal stabilizer leading edge to full-up position.
  (b) Verify PITCH TRIM annunciator extinguishes at the T.O. FWD limit. (See Figure 202, Detail A.)
  (c) Verify PITCH TRIM annunciator extinguishes at the T.O. AFT limit. (See Figure 202, Detail A.)
  (d) Repeat steps E.(4)(a) thru (c) using copilot`s control wheel Trim/Trim Arming Switch.
  (e) Verify all readings fall within the prescribed ranges by trimming from nose up to nose down.
  (f) If readings do not fall within prescribed ranges, perform applicable Functional Test of Horizontal Stabilizer Control System. (Refer to 27-40-00, Adjustment/Test.)

(5) Set Pitch Trim Selector Switch, located on trim switch panel, to SEC.
(6) Set Secondary Pitch Trim Control Switch to NOSE DN, trimming horizontal stabilizer from full-up to full-down position. Set switch to NOSE UP, trimming horizontal stabilizer from full-down to full-up position.
(7) Set Secondary Pitch Trim Control Switch to OFF. With PitchTrim Selector Switch set to SEC, verify stabilizer cannot be trimmed using pilot's or copilot's control wheel Trim/Trim Arming Switches.
(8) Set Pitch Trim Selector Switch to PRI.
(9) Perform Functional Test of Trim-in-Motion (aural warning) System. (Refer to 27-41-00, Adjustment/ Test.)
(10) Set Battery Switches off.





Detail A

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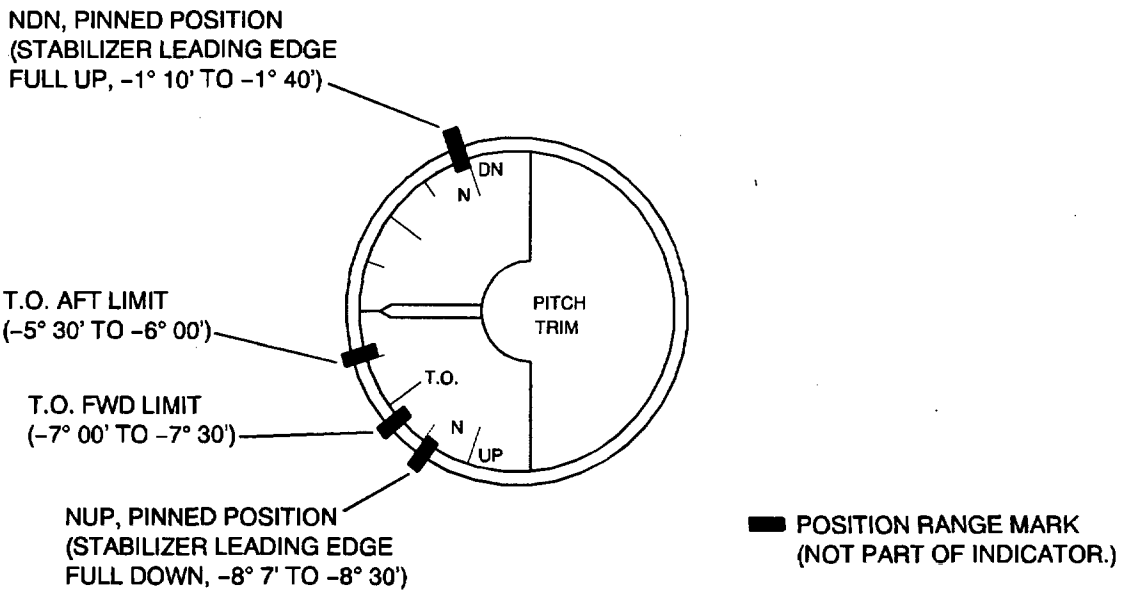
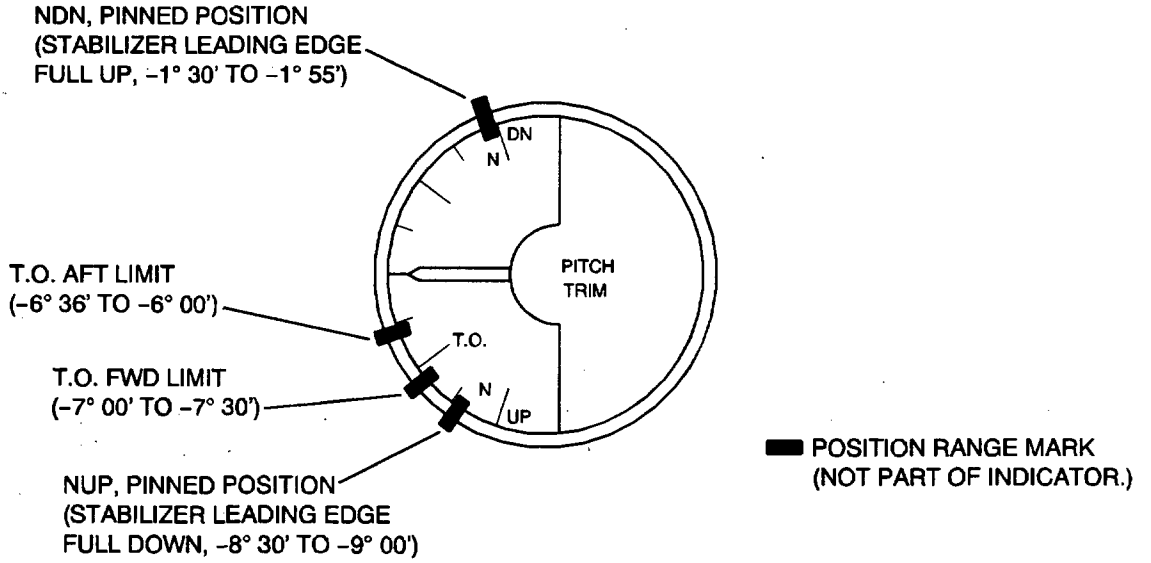
Horizontal Stabilizer Operational Check
Figure 202 (Sheet 1 of 3)

EFFECTIVITY: AIRCRAFT MODIFIED PER SB 35/36-27-21.



35-002 THRU 35-066, 36-002 THRU 36-017, NOT MODIFIED PER AAK 76-4, "REDUCED APPROACH SPEED SYSTEM", AAK 83-2 "INSTALLATION OF FC-530 AUTOPILOT", OR AAK 83-8, "INSTALLATION OF COMPONENTS TO ALLOW VARIABLE PITCH TRIM SETTING AT TAKEOFF".

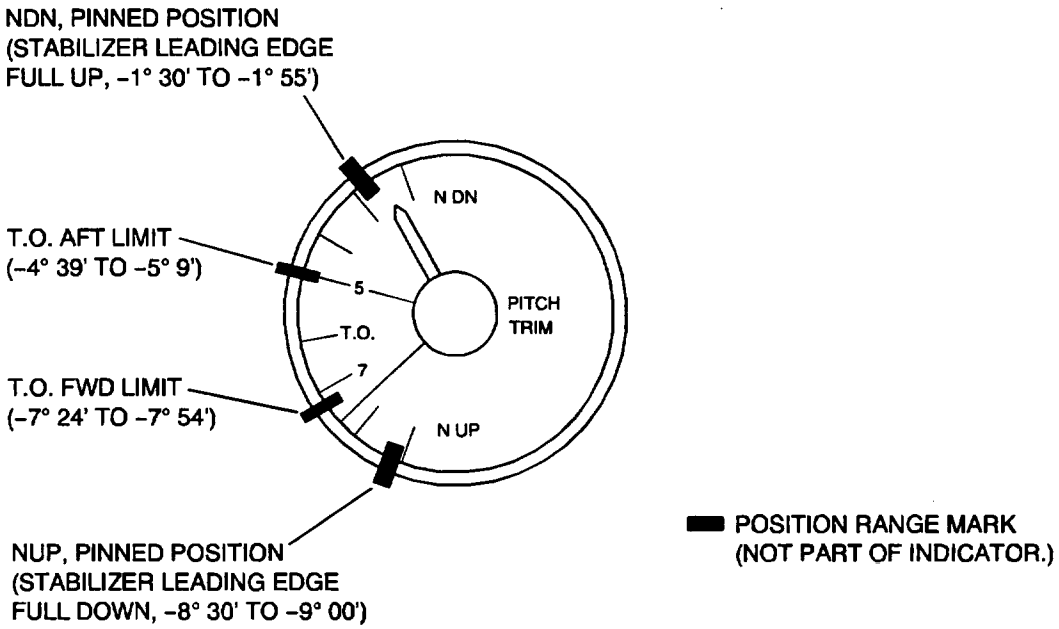


35-067 THRU 35-145, 36-018 THRU 36-035, AND PRIOR AIRCRAFT MODIFIED PER AAK 76-4 "REDUCED APPROACH SPEED SYSTEM" BUT NOT MODIFIED PER AAK 83-2 "INSTALLATION OF FC-530 AUTOPILOT" OR AAK 83-8 " INSTALLATION OF COMPONENTS TO ALLOW VARIABLE PITCH TRIM SETTING AT TAKEOFF".

Detail B

M35-274000-202-02

Horizontal Stabilizer Operational Check
Figure 202 (Sheet 2 of 3)



35-146 AND SUBSEQUENT, 36-036 AND SUBSEQUENT, AND PRIOR AIRCRAFT MODIFIED PER AAK 83-2 "INSTALLATION OF FC-530 AUTOPILOT" OR AAK 83-8 "INSTALLATION OF COMPONENTS TO ALLOW VARIABLE PITCH TRIM SETTING AT TAKEOFF".

Detail B

M35-274000-202-03

Horizontal Stabilizer Operational Check
Figure 202 (Sheet 3 of 3)

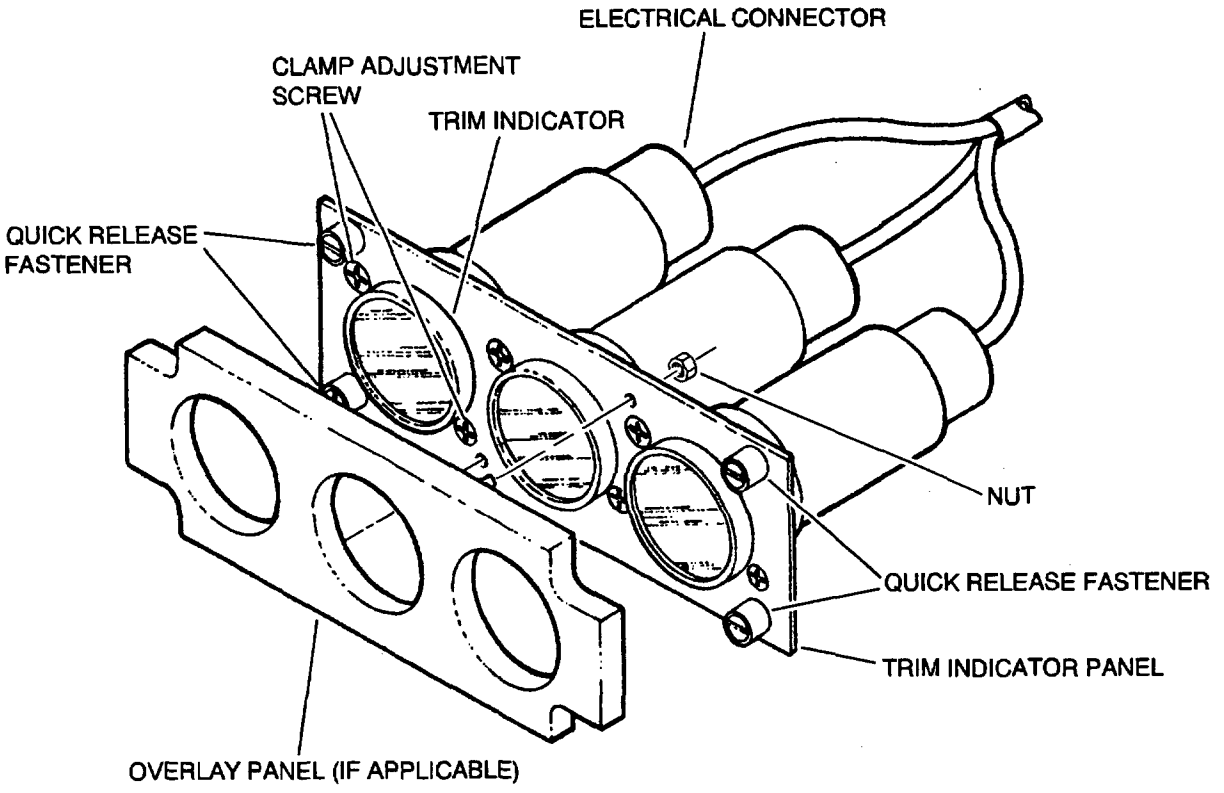
STABILIZER TRIM INDICATOR - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Removal of Stabilizer Trim Indicator (SeeFigure 201.)
      (1) Remove electrical power from aircraft.
      (2) Loosen and remove attaching parts, trim indicator panel from center pedestal.
      (3) Disconnect electrical connector from stabilizer trim indicator.
      (4) Remove attaching parts and stabilizer trim indicator from trim indicator panel.
   B. Installation of Stabilizer Trim Indicator
      (1) Position stabilizer trim indicator in trim indicator panel and secure with attaching parts.
      (2) Connect electrical connector to stabilizer trim indicator.
      (3) Position trim indicator panel in center pedestal. secure with attaching parts.
      (4) Restore electrical power to aircraft.
      (5) Perform Adjustment/Test of stabilizer trim indicator.
2. **Adjustment/Test**
   A. Calibrate stabilizer trim position indicator (See Figure 201.)
      (1) Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

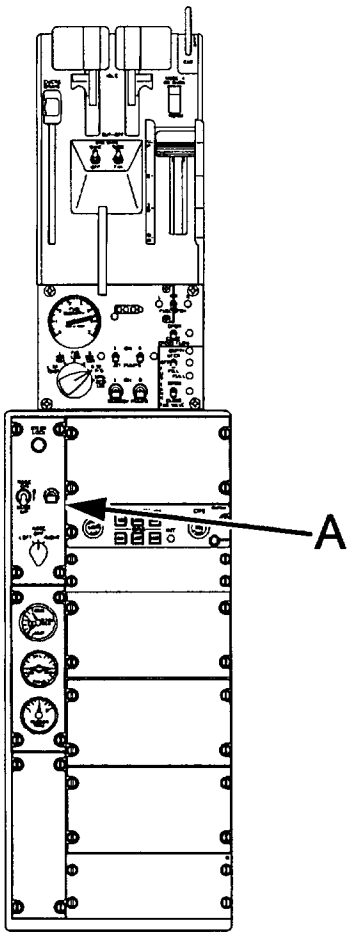
NAME	PART NUMBER	MANUFACTURER	USE
Rigging Pin	3170037-16	Learjet Corp. Wichita, KS	Rigging.

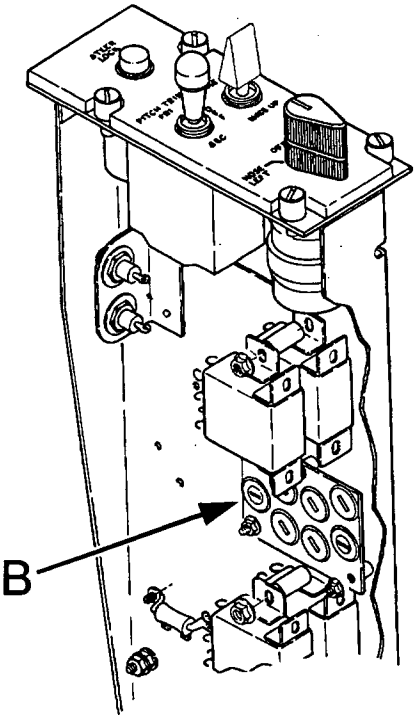
      (2) Remove equipment as necessary from the pedestal adjacent to the trim switch panel. (Refer to 31-11-00.) This allows access to the position indicator calibration assembly.
      (3) Remove upper vertical stabilizer access panel.
      (4) Actuate horizontal stabilizer to full up position.
      (5) *On Aircraft 35-001 thru 35-676 and 36-001 thru 36-063 modified per SB 35/36-27-21,* insert rigging pin in rigging bracket and through hole in rigging angle located on horizontal stabilizer assembly. (Refer to 27-40-00, Inspection/Check.)
      (6) Adjust nose down potentiometer to obtain full nose down indication on stabilizer trim indicator.
      (7) Actuate horizontal stabilizer to full down position.
      (8) Adjust nose up potentiometer to obtain full nose up indication on stabilizer trim indicator.
      (9) Cycle horizontal stabilizer between full up and full down positions while checking respective indications on stabilizer trim indicator. Readjust Nose Up and Nose Down potentiometers as necessary to obtain coordinated indications.
      (10) Replace upper vertical stabilizer access panel.
      (11) Replace and secure equipment in pedestal.



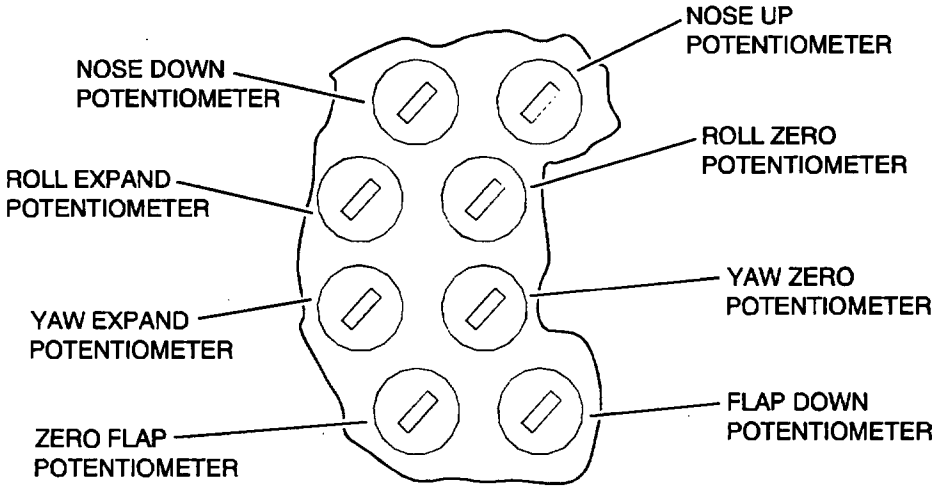
M35-274001-201-01

Stabilizer Trim Indication Installation
Figure 201





Detail A



Detail B

M35-274001-202-01

Position Indicator Calibration Assembly
Figure 202

PITCH TRIM SPEED CONTROLLER - MAINTENANCE PRACTICES

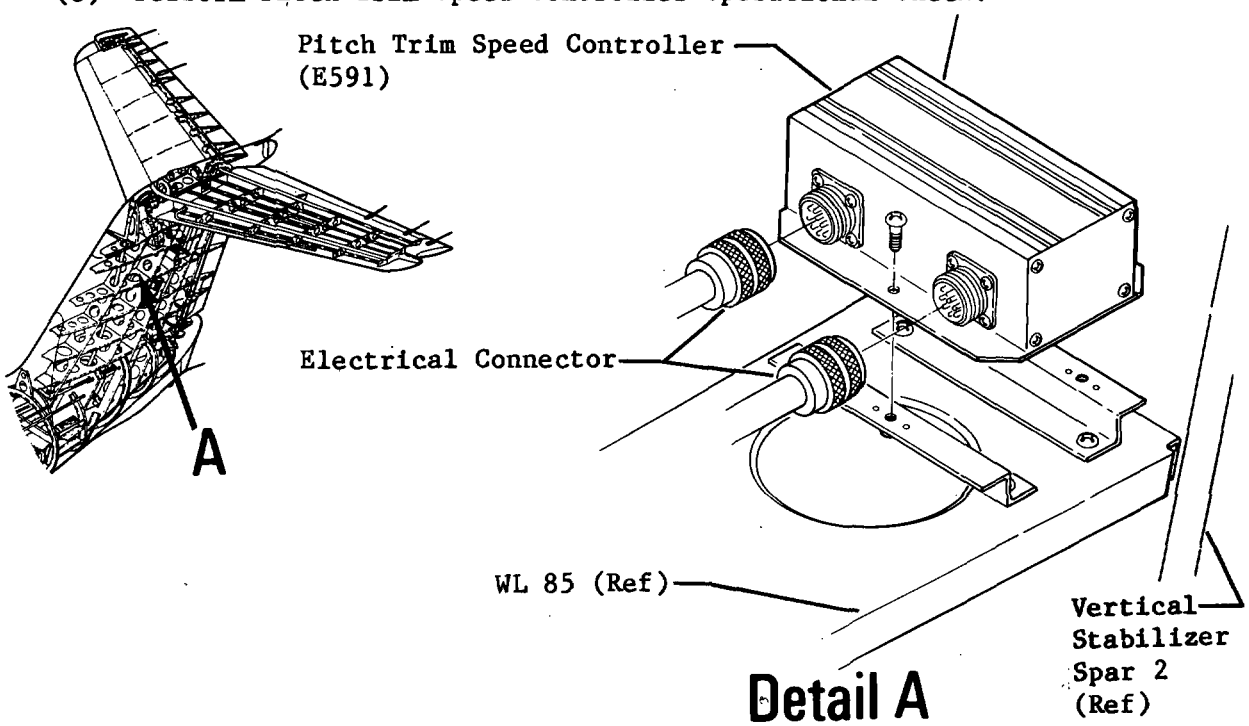
1. REMOVAL/INSTALLATION

A. Remove Pitch Trim Speed Controller (See figure 201.)
   (1) Remove electrical power from aircraft.
   (2) Remove vertical stabilizer access panel located on RH side between WL 85 and WL 95 and between spars 1 and 2 to gain access to pitch trim speed controller.
   (3) Disconnect electrical connectors from pitch trim speed controller.
   (4) Remove attaching parts and pitch trim speed controller assembly from aircraft.

B. Install Pitch Trim Speed Controller (See figure 201.)
   (1) Position pitch trim speed controller in vertical stabilizer and secure with attaching parts.

   NOTE: Install bonding jumper between pitch trim speed controller box frame and aircraft structure. Check electrical resistance as instructed in Chapter 20 of the Wiring Manual.

   (2) Connect electrical connectors to pitch trim speed controller.
   (3) Install and secure vertical stabilizer access panel.
   (4) Restore electrical power to aircraft.
   (5) Perform Pitch Trim Speed Controller Operational Check.



Pitch Trim Speed Controller Installation
Figure 201

2. **INSPECTION/CHECK**

A. Pitch Trim Speed Controller Operational Check

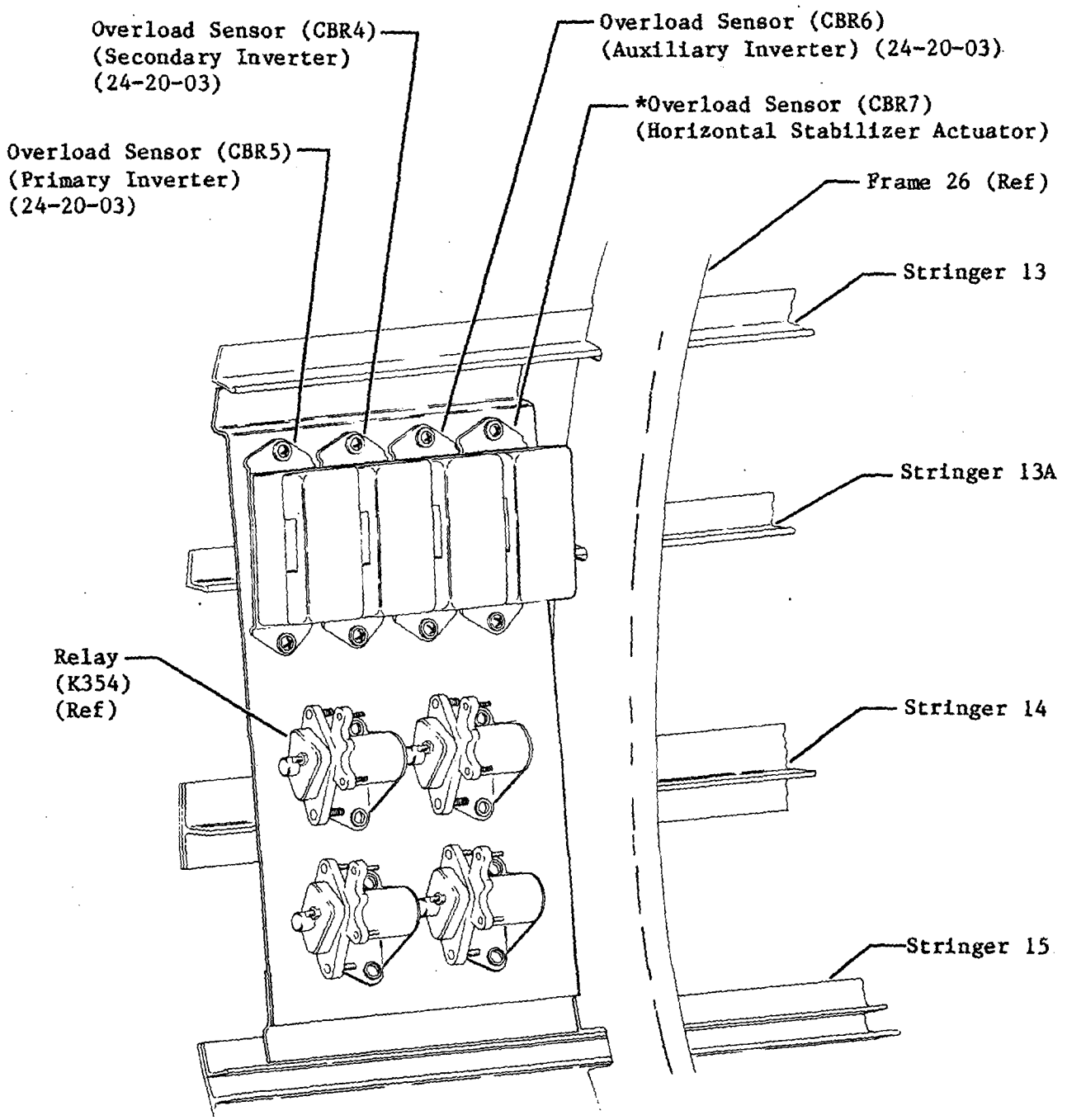
(1) Apply 28 vdc auxiliary power to aircraft.
(2) Set Battery Switches to BAT 1 and BAT 2.
(3) Set Pitch Trim Switch to PRI.
(4) Position flaps up.
(5) Measure elapsed time as stabilizer moves from up stop to down stop using either pilot's or copilot's Trim and Trim Arming Switch.
(6) Stop to stop elapsed time shall be 65 to 69 seconds.
(7) Position flaps down.
(8) Measure elapsed time as stabilizer moves from up stop to down stop using either Pilot's or Copilot's Trim and Trim Arming Switch.
(9) Stop to stop elapsed time shall be 14 to 20 seconds.
(10) Set Pitch Trim Switch to SEC.
(11) Actuate Secondary Pitch Trim Switch while measuring elapsed time from stop to stop.
(12) Stop to stop elapsed time shall be less than 68 seconds.
(13) Operate trim from either pilot's or copilot's side while raising flaps simultaneously. Trim speed shall change from high speed to low speed as flaps retract through 3°.
(14) Position flaps up.
(15) Set Test Selector Switch to TRIM OVSP.
(16) Set Pitch Trim Switch to PRI and depress Test Switch pushbuttons. Verify that PITCH TRIM warning light on glareshield panel illuminates while operating the pitch trim system.
(17) Release Test Switch pushbutton and set Test Selector Switch to TRIM MON. Depress Test Switch pushbutton and verify that PITCH TRIM warning light illuminates.
(18) Release Test Switch pushbutton and verify that PITCH TRIM warning light extinguishes.
(19) Remove 28 vdc auxiliary power from aircraft.
(20) Set Battery Switches to OFF.

HORIZONTAL STABILIZER ACTUATOR OVERLOAD SENSOR - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

A. Remove Horizontal Stabilizer Actuator Overload Sensor (See figure 201.)
   (1) Remove electrical power from aircraft.
   (2) Lower tailcone access door.
   (3) Remove attaching parts and cover from sensor.
   (4) Disconnect electrical wiring from sensor.
   (5) Remove attaching parts and sensor from aircraft.

B. Install Horizontal Stabilizer Actuator Overload Sensor (See figure 201.)
   (1) Install sensor and secure with attaching parts.
   (2) Connect electrical wiring to sensor.
   (3) Install cover and secure with attaching parts.
   (4) Restore electrical power to aircraft.
   (5) Perform pitch trim system operational check. (Refer to 27-40-00.)
   (6) Raise and secure tailcone access door.



Horizontal Stabilizer Actuator Overload Sensor Installation
Figure 201

9-96BW

HORIZONTAL STABILIZER - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Hoist & Sling	2331001	Learjet Inc. Wichita, KS	Support stabilizer.
Driver	MIT 2331028-4	Learjet Inc. Wichita, KS	Remove/install hinge pin.

2. Removal/Installation

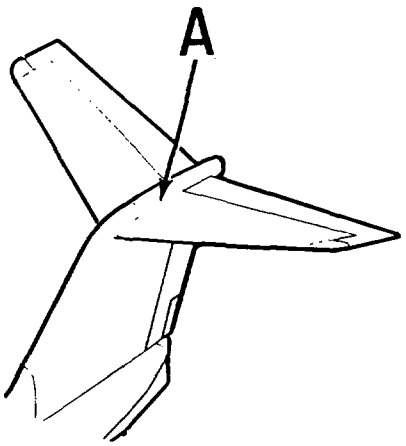
NOTE:
- The horizontal stabilizer is attached to the vertical stabilizer at a single hinge point and at the horizontal stabilizer actuator.
- Access to the horizontal stabilizer hinge pin is through access covers on the bottom side of each stabilizer half.

A. Remove Horizontal Stabilizer (See Figure 201.)
(1) Set Battery Switches on.
(2) Depress and hold pilot's Trim and Trim Arming Switch to nose up. When horizontal stabilizer has reached the extreme down position, release Trim and Trim Arming Switch.
(3) Set Battery Switches off.
(4) Remove elevators. (Refer to Chapter 27.)
(5) Remove rib fairing assembly, bulkhead fairing assembly, and forward fin tip fairing from vertical stabilizer. Remove hinge pin access covers from RH and LH sides of horizontal stabilizer. Disconnect bonding jumper from horizontal stabilizer. Leave opposite end connected to vertical hinge fittings.
(6) Disconnect stabilizer anti-ice plumbing. (Refer to Chapter 30.)
(7) Remove attaching parts and elevator downspring push-pull tube(s) from horizontal stabilizer.
(8) Remove screws from upper surface of stabilizer. The screws are located outboard of BL 19 on both sides. Attach hoist and support sling to stabilizer using NAS220-15 screws.
(9) Remove attaching parts from stabilizer actuator and stabilizer attach point.
(10) Remove safety wire, bolt, and stop from hinge pin.
(11) Remove hinge pin using 5X rivet gun and driver.

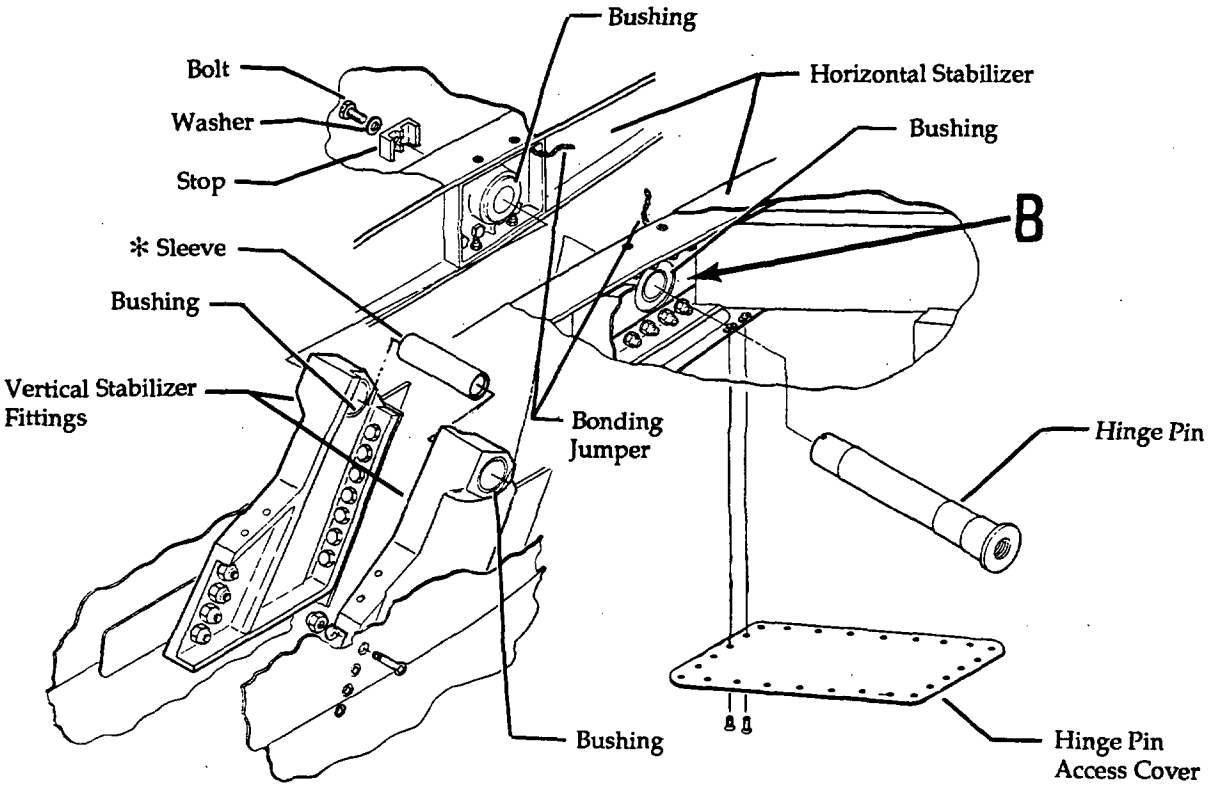
CAUTION: USE CARE WHEN REMOVING HORIZONTAL STABILIZER FROM VERTICAL STABILIZER OR DAMAGE TO RIGGING PLATE MAY OCCUR.

(12) Remove stabilizer from aircraft.
(13) Inspect phenolic coating on stabilizer bushings. If phenolic coating is worn through at any point, replace bushings. (Refer to Chapter 55.)

B. Install Horizontal Stabilizer (See Figure 201.)
(1) Clean hinge pin and place in alcohol/dry ice mixture for a minimum of 20 minutes prior to installation.

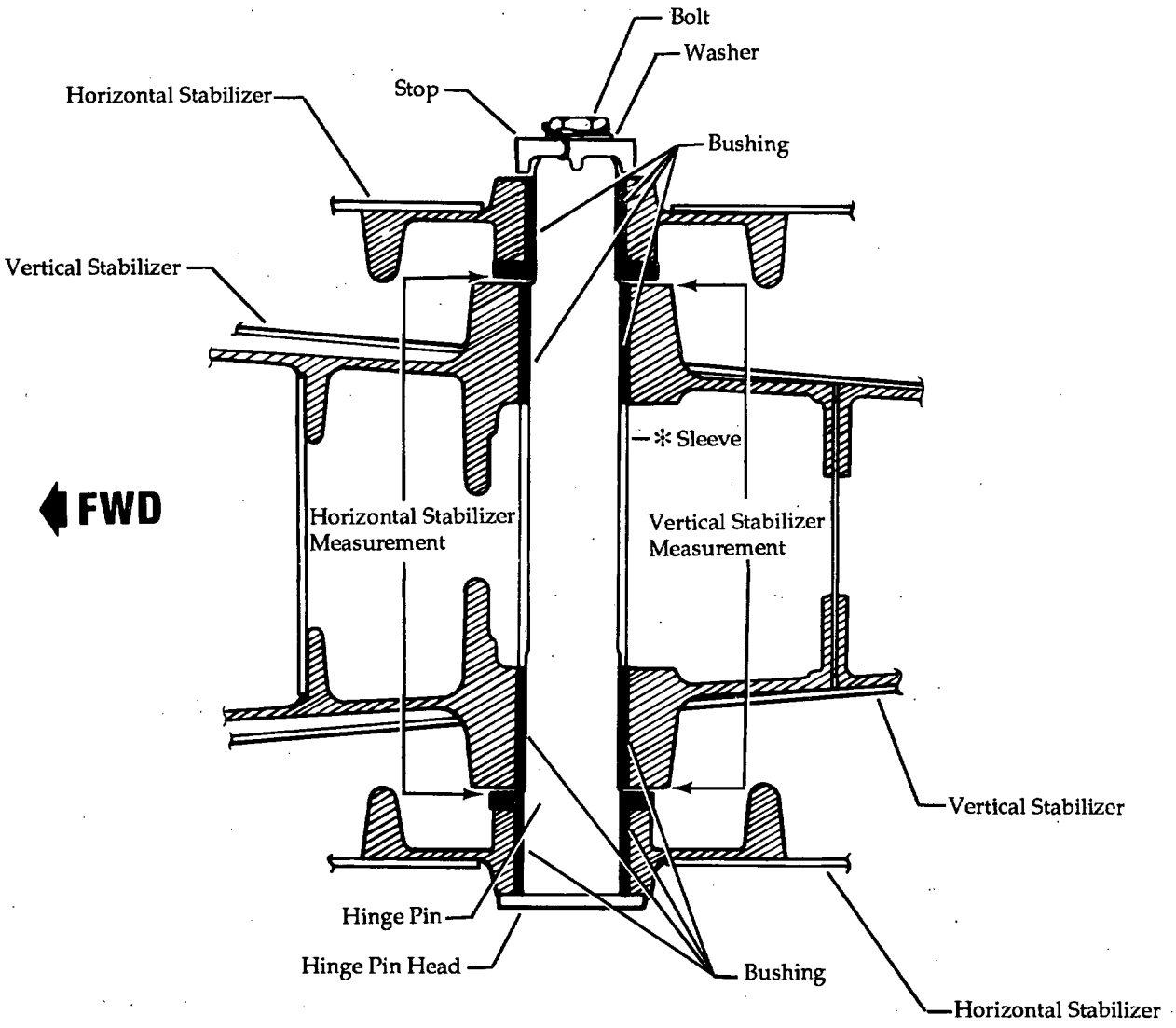


* *Effective 35-303 and Subsequent, 36-045 and Subsequent and Prior Aircraft Modified Per SB 35/36-27-13, Add Sleeve as Shown.*



Detail A

Horizontal Stabilizer Installation
Figure 201 (Sheet 1 of 2)



(VIEW LOOKING DOWN)

Detail B

Horizontal Stabilizer Installation
Figure 201 (Sheet 2 of 2)

(2) Measure distance between horizontal stabilizer bushing faces (inside measurement) and record measurement. (See Figures 201 and 202.)
(3) Measure width across vertical stabilizer hinges (outside measurement) and record measurement. (See Figures 201 and 202.)

CAUTION: TOTAL GAP MUST BE BETWEEN 0.001 TO 0.07 INCH (0.0254 TO 01.778 MM).

IF THIS GAP IS NOT ATTAINED CONTACT LEARJET FIELD SERVICE.

(4) Calculate total gap by subtracting vertical stabilizer hinge width from horizontal stabilizer bushing distance.
(5) Calculate thickness of temporary assembly shim by dividing gap by 2. (See Figure 202.)
(6) Position horizontal stabilizer so that holes in vertical stabilizer fittings and horizontal stabilizer hinge fittings are aligned.

NOTE: On *Aircraft 35-303 and Subsequent, 36-045 and Subsequent, and prior aircraft modified per SB 35/36-27-13*, a sleeve is installed between the vertical stabilizer hinge fittings and over the stabilizer hinge pin. The sleeve must be properly aligned and held in place while the hinge pin is driven in place.

(7) Insert temporary assembly shim between the left side of the vertical stabilizer hinge fitting and the left horizontal stabilizer bushing face.

WARNING: SKIN CONTACT WITH ALCOHOL/DRY ICE MIXTURE OR COLD HINGE PIN CAN RESULT IN PERSONAL INJURY. UTILIZE SUITABLE TOOLS FOR HANDLING COLD HINGE PIN.

(8) Remove hinge pin from alcohol/dry ice container and insert in aligned holes. Drive hinge pin into place (head of hinge pin flush with horizontal stabilizer pivot fitting) using 5X rivet gun and driver. Hinge pin must be driven in place within a maximum of 14 seconds after removal from alcohol/dry ice mixture.
(9) Remove temporary assembly shim from left side of vertical stabilizer hinge fitting.
(10) Install stop on hinge pin and secure with bolt. Tighten bolt finger-tight.
(11) Measure clearance between stop and horizontal stabilizer pivot fitting.
(12) Remove bolt and stop from hinge pin.
(13) Remove material from stop to achieve clearance between stop and right horizontal stabilizer pivot fitting as calculated on worksheet. (See Figure 202.)
(14) Paint bare aluminum portion of stop with epoxy primer.
(15) Install stop, washer and bolt on hinge pin. Tighten bolt and verify stop to horizontal stabilizer pivot fitting clearance.
(16) Torque bolt to 160 to 190 inch pounds and safety wire.
(17) Secure bonding jumper to stabilizer with attaching parts. (Refer to Wiring Manual, Chapter 20.)
(18) Secure stabilizer actuator to stabilizer with attaching parts.
(19) Install elevators. (Refer to 27-30-03.) Do not install fairings at this time.
(20) Position elevator downspring push-pull tube(s) up to horizontal stabilizer, and secure with attaching parts.
(21) Connect stabilizer anti-ice plumbing. (Refer to 30-10-02.)
(22) Install bulkhead fairing assembly, rib fairing assembly, forward tip fairing, tip access fairing, aft tip fairing stabilizer fairings, and hinge pin access panels.
(23) Check clearances of fairings. (Refer to 27-30-03.)
(24) Rig horizontal stabilizer. (Refer to 27-40-00.)
(25) Rig elevator control system. (Refer to 27-30-00.)
(26) Restore aircraft to normal.

HORIZONTAL STABILIZER INSTALLATION WORKSHEET

B.(2) HORIZONTAL STAB. BUSHING DISTANCE = _____

B.(3) VERTICAL STABILIZER HINGE WIDTH = _____

TOTAL GAP B.(2) - B.(3) = _____ (TOTAL GAP MUST BE BETWEEN 0.001 TO 0.07 INCHES. IF GAP IS NOT ATTAINED, CONTACT LEARJET FIELD SERVICE)

B.(4) SHIM THICKNESS = GAP/2 ± 0.010 inch

_____ / 2 = _____ ± 0.010 inch

B.(11) MINIMUM STOP CLEARANCE = SHIM THICKNESS _____

+ 0.015 inch

= _____

MAXIMUM STOP CLEARANCE = SHIM THICKNESS _____

+ 0.020 inch

= _____

Horizontal Stabilizer Installation Worksheet
Figure 202

HORIZONTAL STABILIZER ACTUATOR - MAINTENANCE PRACTICES

1. **Removal/Installation**

A. Removal of Horizontal Stabilizer Actuator (*Aircraft 35-002 thru 35-041 and 36-002 thru 36-016*) (See Figure 201.)

(1) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.

(2) Remove forward vertical stabilizer tip access cover and stabilizer actuator access cover from vertical stabilizer.

WARNING: WHEN THE STABILIZER ACTUATOR IS DISCONNECTED FROM THE HORIZONTAL STABILIZER (WITH ELEVATORS INSTALLED), THE STABILIZER WILL ABRUPTLY SHIFT TO THE FULL-UP POSITION. CARE SHOULD BE TAKEN TO ENSURE NO PHYSICAL OR STRUCTURAL DAMAGE WILL OCCUR.

(3) Disconnect electrical connector from actuator.

(4) Remove attaching parts securing actuator to horizontal stabilizer.

(5) Remove attaching parts securing actuator to vertical stabilizer spar. Remove actuator from aircraft.

B. Installation of Horizontal Stabilizer Actuator (*Aircraft 35-002 thru 35-041 and 36-002 thru 36-016*) (See Figure 201.)

(1) Attach actuator lower fitting to vertical stabilizer spar and secure with attaching parts.

(2) Attach actuator upper fitting to horizontal stabilizer and secure with attaching parts.

(3) Connect electrical connector to actuator.

(4) On *Aircraft not modified per SB 35/36-27-21*, rig horizontal stabilizer. (Refer to 27-40-00, Adjustment/Test.)

(5) On *Aircraft 35-002 thru 35-676 and 36-002 thru 36-063 modified per SB 35/36-27-21*, perform Operational Check of Horizontal Stabilizer Control (pitch trim) system. (Refer to 27-40-00, Inspection/Check.)

(6) Install forward vertical stabilizer tip access cover and stabilizer actuator cover on vertical stabilizer.

(7) Install horizontal stabilizer fairings on both sides of vertical stabilizer.

(8) Check clearance between horizontal stabilizer fairing and vertical stabilizer. Clearance should be 0.0 to 0.13 inch (0 to 3.3 millimeters).

C. Removal of Horizontal Stabilizer Actuator (*Aircraft 35-042 and Subsequent, 36-017 and Subsequent*) (See Figure 201.)

(1) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.

(2) Remove forward vertical stabilizer tip access cover and stabilizer actuator access cover from vertical stabilizer.

(3) Disconnect electrical connectors from actuator.

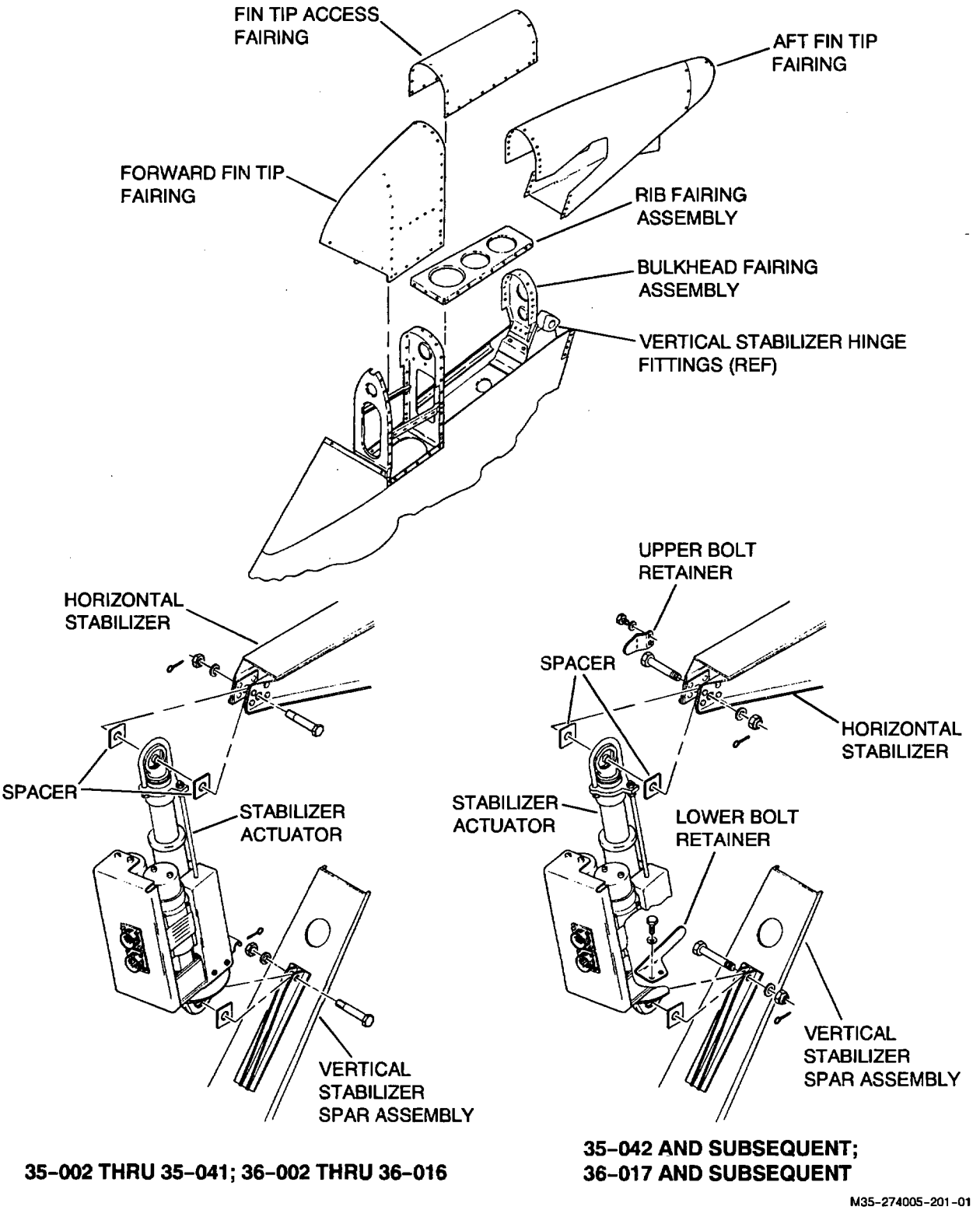
WARNING: WHEN THE STABILIZER ACTUATOR IS DISCONNECTED FROM THE HORIZONTAL STABILIZER (WITH ELEVATORS INSTALLED), THE STABILIZER WILL ABRUPTLY SHIFT TO THE FULL-UP POSITION. CARE SHOULD BE TAKEN TO ENSURE NO PHYSICAL OR STRUCTURAL DAMAGE WILL OCCUR.

(4) Remove attaching parts and upper actuator bolt retainer.

(5) Remove attaching parts securing actuator to horizontal stabilizer.

(6) Remove attaching parts and lower actuator bolt retainer.
(7) Remove attaching parts securing actuator ground strap.
(8) Remove attaching parts securing actuator to vertical stabilizer spar and remove actuator from aircraft.

D. Installation of Horizontal Stabilizer Actuator (*Aircraft 35-042 and Subsequent, 36-017 and Subsequent,*) (See figure 201.)
(1) Install actuator and secure lower fitting to vertical stabilizer spar with attaching parts.
(2) Install lower bolt retainer and secure with attaching parts.
(3) Secure actuator ground strap to vertical stabilizer spar with attaching parts.
(4) Secure upper actuator fitting to horizontal stabilizer with attaching parts.
(5) Install upper bolt retainer and secure with attaching parts.
(6) Connect electrical connectors to actuator.
(7) On *Aircraft not modified per SB 35/36-27-21*, rig horizontal stabilizer. (Refer to 27-40-00, Adjustment/Test.)
(8) On *Aircraft 35-002 thru 35-676 and 36-002 thru 36-063 modified per SB 35/36-27-21*, perform Operational Check of Horizontal Stabilizer Control (pitch trim) system. (Refer to 27-40-00, Inspection/Check.)
(9) Install forward vertical stabilizer tip access cover and stabilizer actuator access cover on vertical stabilizer.
(10) Install horizontal stabilizer fairing on both sides of vertical stabilizer.
(11) Check clearances between horizontal stabilizer fairing and vertical stabilizer. Clearance should be 0.0 to 0.13 inch (0.0 to 3.3 millimeters).



Horizontal Stabilizer Actuator Installation
Figure 201

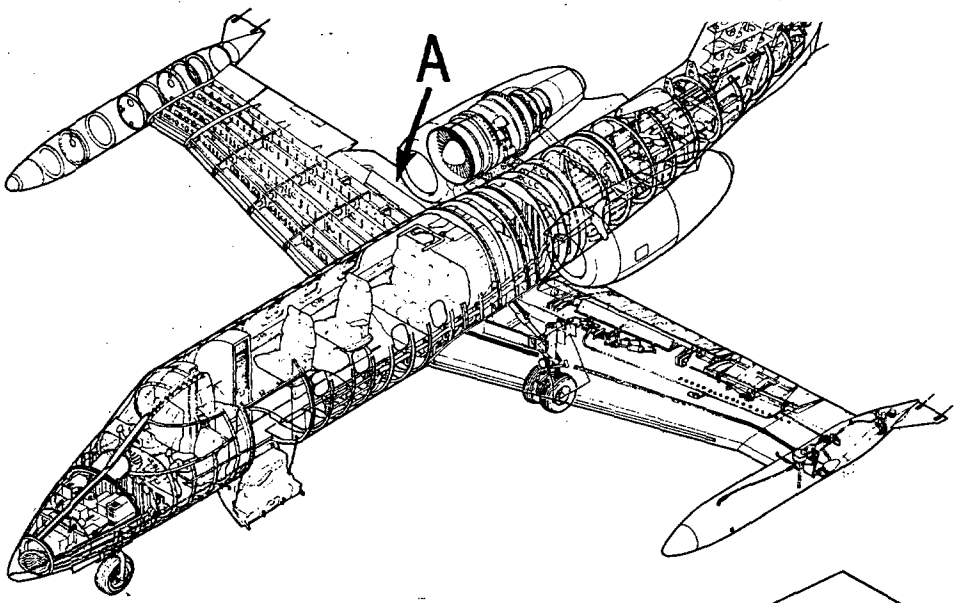
TRIM SPEED SWITCH - MAINTENANCE PRACTICES

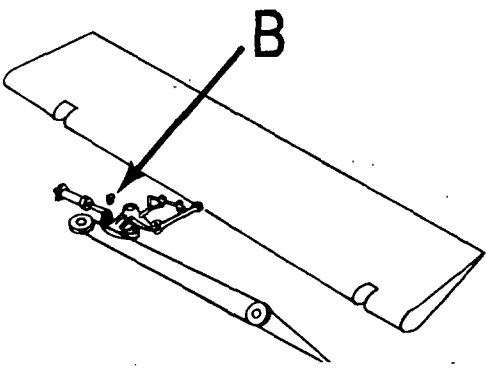
1. **Removal/Installation**
   A. Remove Trim Speed Switch (S295). (See Figure 201.)
      (1) Remove RH wing access panel between spar 7 and spar 8 to expose RH flap sector and trim speed switch.
      (2) With flaps in the full up position measure from bottom of trim speed switch bracket (not mounting plate) to flat surface of flap sector. Record measurement.
      (3) Connect external hydraulic power to aircraft.
      (4) Connect external electrical power to aircraft.
      (5) Set battery switches on.
      (6) Lower flaps.
      (7) Set battery switches off and remove external electrical power from aircraft.
      (8) Disconnect aircraft batteries.
      (9) Disconnect and identify electrical wiring from trim speed switch.
      (10) Remove attaching parts securing switch, shim, and mounting plate to bracket.
      (11) Remove switch, shim, and mounting plate from aircraft.
      (12) When installing a replacement switch, disassemble switch from mounting plate.
   B. Install Trim Speed Switch (S295). (See Figure 201.)
      (1) When installing a replacement trim speed switch, install switch to mounting plate using spacer, key washer, and nut.
      (2) Align axis of roller with keyway and tighten roller guide. Ensure that roller is free.
      (3) Measure distance from top of mounting plate to end of relaxed switch.
      (4) Add shims as required (two maximum) until distance from top of shims to bottom of relaxed switch is 0.100 inch greater than distance from bottom of switch bracket to top of flap sector (measured prior to switch removal).
      (5) Position switch, mounting plate, and shims on bracket and secure with attaching parts.
      (6) Safety wire hex head nut.
      (7) Identify and connect switch electrical wiring to aircraft wiring. (Refer to Wiring Manual.)
      (8) Connect aircraft batteries.

      CAUTION: DO NOT ATTEMPT TO ACTUATE FLAPS UNTIL SWITCH IS ADJUSTED OR SWITCH BRACKET ASSEMBLY WILL BE DAMAGED.

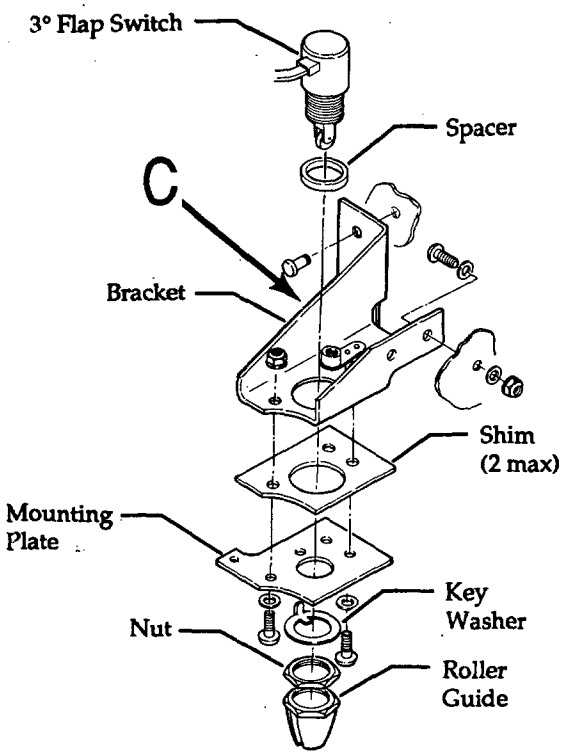
      (9) Perform functional check of trim speed switch. (Refer to Inspection/Check, this section.)
      (10) Install RH wing access panel.

2. **Inspection/Check**
   A. Functional Check of Trim Speed Switch (S295).
      (1) Connect external electrical power to aircraft.
      (2) Set Battery Switches on.
      (3) Set Pitch Trim Switch on pedestal to PRI.
      (4) Using pilot's or copilot's Trim and Trim Arming Switch, trim horizontal stabilizer to either stop.
      (5) Ensure flaps are at full down position.
      (6) Using pilot's or copilot's Trim and Trim Arming Switch, trim horizontal stabilizer to opposite stop while raising flaps simultaneously. Note the rate of trim change from high speed to low speed when flaps are fully retracted.
      (7) Set Battery Switches off.
      (8) Remove external electrical power from aircraft.



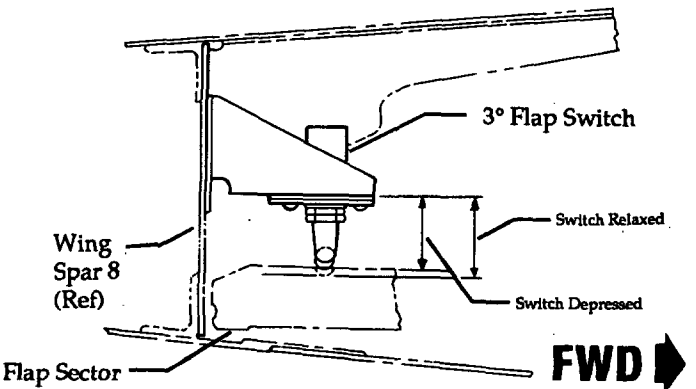


Detail A



FWD

Detail B



Detail C

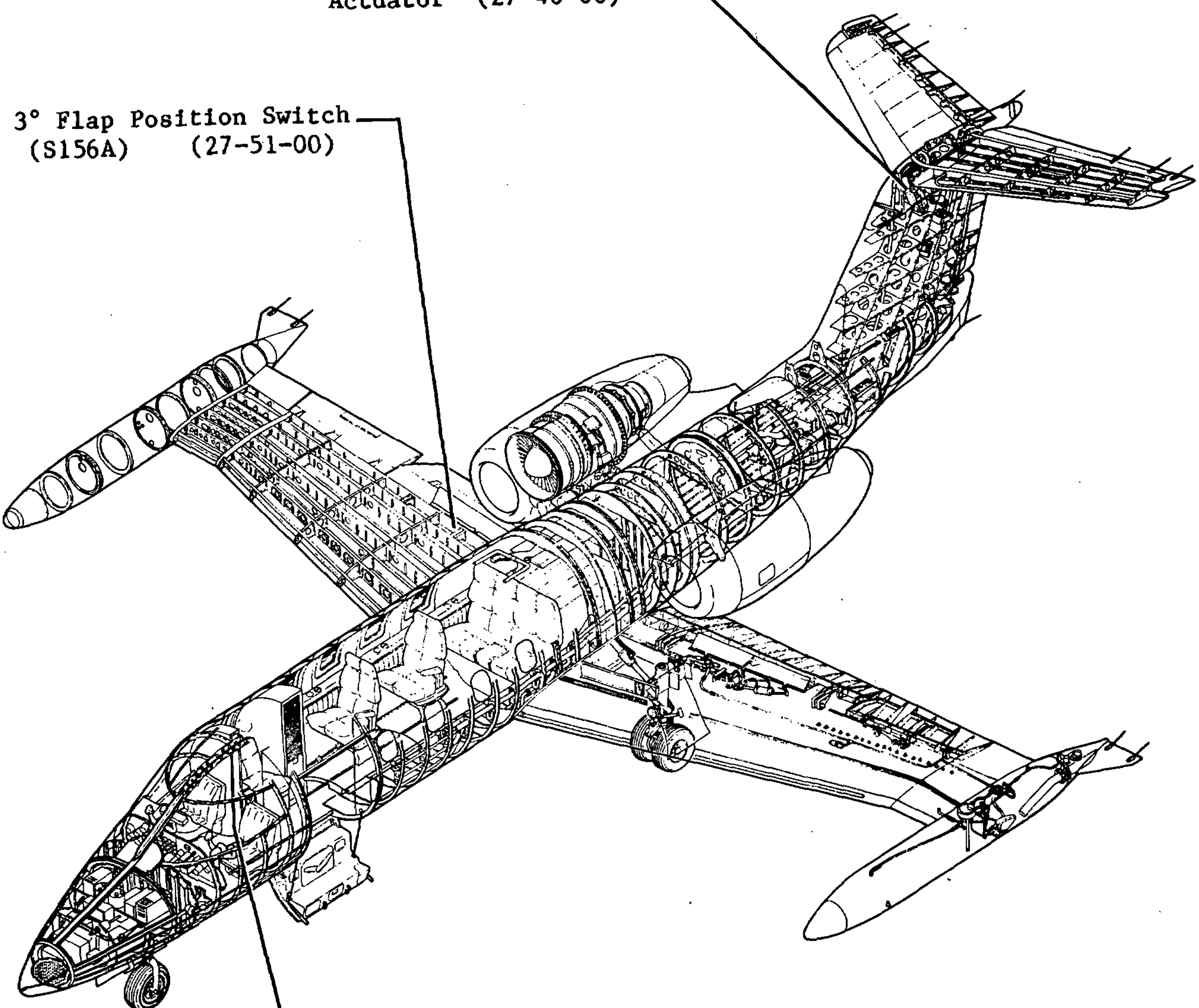
A8-50B
BC9-307C

Trim Speed Switch Installation
Figure 201

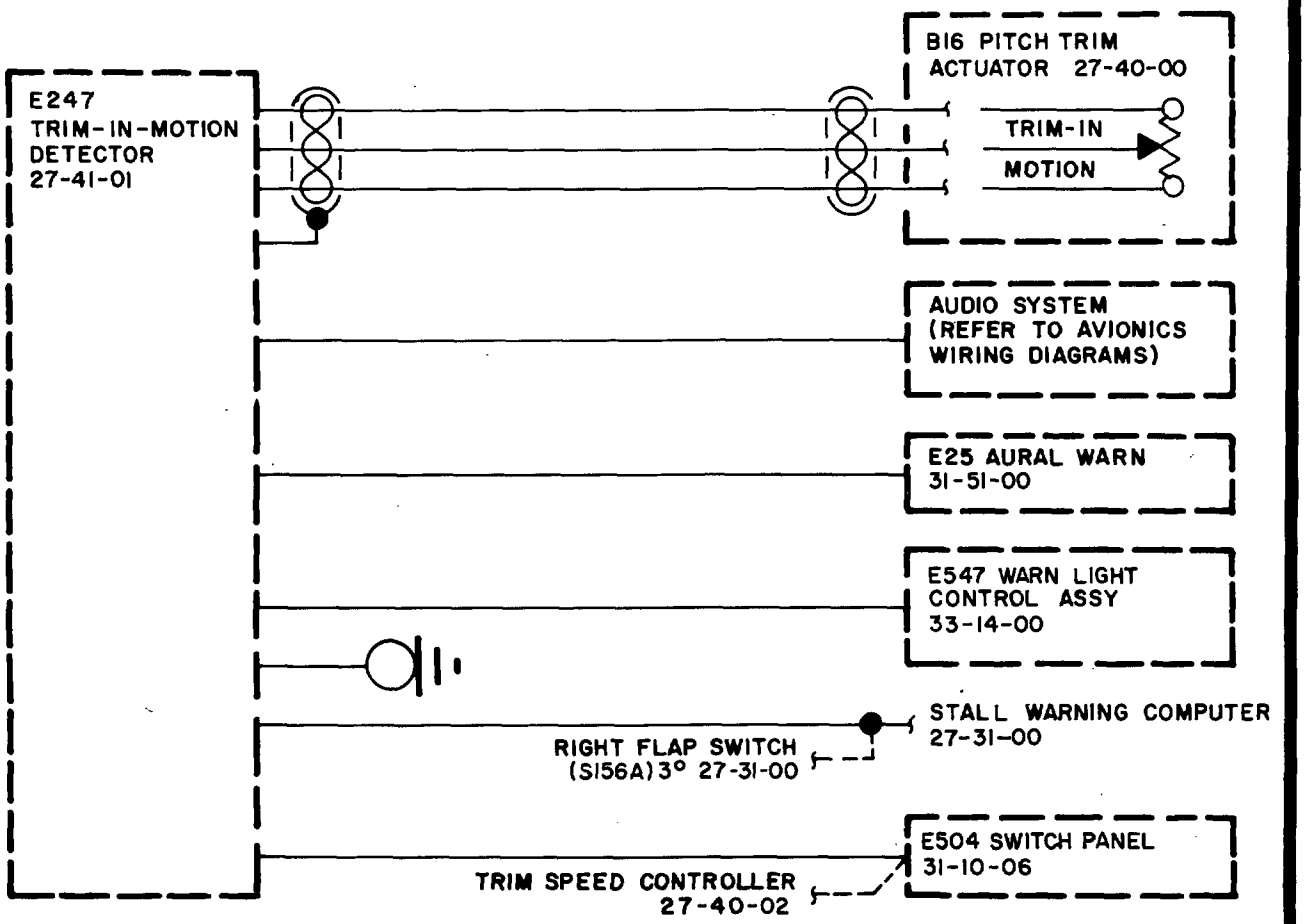
TRIM-IN-MOTION SYSTEM - DESCRIPTION AND OPERATION

1. **DESCRIPTION** (See figure 1.)
   A. The trim-in-motion system alerts the crew to pitch trim motion (with flaps up) by sounding a warning (cricket) through the cockpit speaker and headphone audio systems.
   B. The trim-in-motion system components are a trim-in-motion system detector box (E247),a linear potentiometer, and a 3° flap switch (S156A) (27-50-05).
   C. **Component Description**
      (1) On Aircraft 35-002 thru 35-505 and 36-002 thru 36-053 modified per AAK 83-2, "Installation of FC-530 Autopilot," the trim-in-motion system detector box is located behind the upholstery on the RH side of the cockpit forward of frame 10 between stringers 14 and 15. On Aircraft 35-506 and Subsequent, 36-054 and Subsequent, the trim-in-motion system detector box is located behind the copilot's seat aft of frame 9.
      (2) The linear potentiometer is part of the horizontal stabilizer actuator.

2. **OPERATION** (See figure 2.)
   A. Horizontal stabilizer actuator movement (pitch trim) modulates signal output of the actuator linear potentiometer which acts on the trim-in-motion detector box circuitry. The trim-in-motion detector box signals the audible tone generator which produces a series of clicks heard through the cockpit speakers and headphones. Audio clicking (cricket) shall sound with virtually no delay after the horizontal stabilizer trimming action starts.
   B. The trim-in-motion system (aural warning) is inoperative when the flaps are extended beyond 3°.



Trim-in-Motion Component Location
Figure 1



Trim-in-Motion System Electrical Schematic
Figure 2

TRIM-IN-MOTION SYSTEM - MAINTENANCE PRACTICES

1. Adjustment/Test

A. Trim-in-Motion System Volume Adjustment (See Figure 201.)

NOTE: It is recommended that volume adjustment be accomplished so that the clicker (cricket) is audible at a high altitude, high-speed cruise (maximum normal noise levels).

(1) Ensure that flaps are fully retracted.
(2) Activate either the primary or secondary pitch trim system.
(3) Operate pitch trim and adjust potentiometer on trim-in-motion detector box to vary volume.

NOTE: Adjust the aft potentiometer for volume control.

2. Inspection/Check

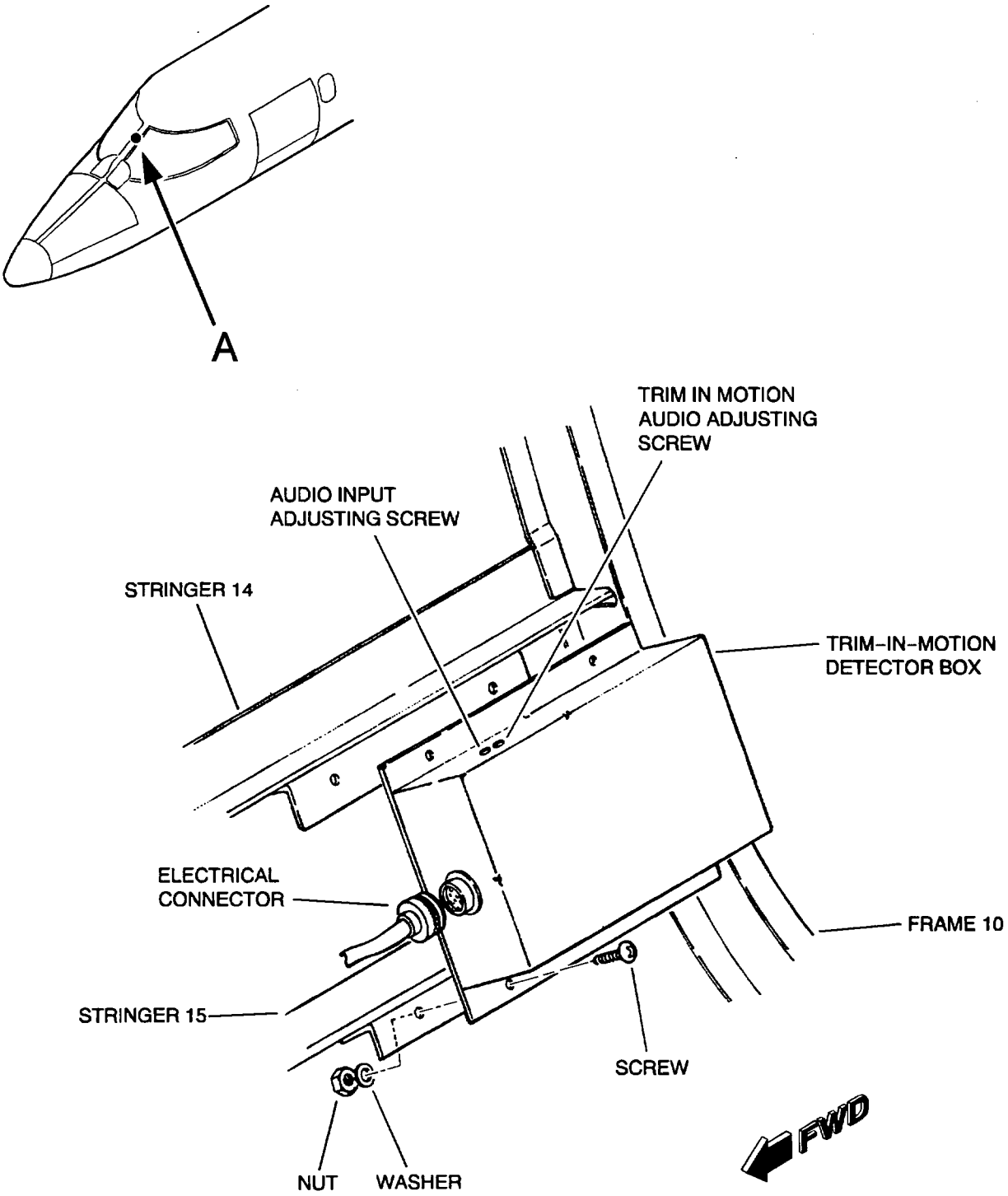
A. Trim-in-Motion System Operational Check *(Aircraft 35-002 thru 35-505, 36-002 thru 36-053 not modified per AAK 83-2.)*

(1) Connect external electrical power to aircraft.
(2) Set Battery Switches on.
(3) Ensure that flaps are fully retracted.
(4) Set Pitch Trim Selector Switch to PRI.
(5) Set and hold pilot's Trim and Trim Arming Switch to NOSE UP. Horizontal stabilizer leading edge shall lower and audio clicking (cricket) shall sound.
(6) Release pilot's Trim and Trim Arming Switch. Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(7) Set and hold pilot's Trim and Trim Arming Switch to NOSE DN. Horizontal stabilizer leading edge shall raise and audio clicking (cricket) shall sound.
(8) Depress pilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(9) Release pilot's Control Wheel Master Switch. Horizontal stabilizer trimming and audio clicking shall resume.
(10) Depress copilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(11) Release copilot's Control Wheel Master Switch. Horizontal stabilizer movement and audio clicking shall resume.
(12) Repeat steps (5) thru (11) using copilot's Trim and Trim Arming Switch.
(13) Set Pitch Trim Selector Switch to OFF.
(14) Set pilot's Trim and Trim Arming Switch to NOSE UP. No horizontal stabilizer movement or audio clicking (cricket) shall occur.
(15) Set pilot's Trim and Trim Arming Switch to NOSE DN. No horizontal stabilizer movement or audio clicking (cricket) shall occur.
(16) Repeat steps (14) and (15) using copilot's Trim and Trim Arming Switch.
(17) Set Pitch Trim Selector Switch to SEC.
(18) Set Secondary Pitch Trim Switch to NOSE UP. Horizontal stabilizer leading edge shall lower and audio clicking (cricket) shall sound.
(19) Release Secondary Pitch Trim Switch. Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(20) Set and hold Secondary Pitch Trim Switch to NOSE DN. Horizontal stabilizer leading edge shall raise and audio clicking (cricket) shall sound.
(21) Depress pilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(22) Release pilot's Control Wheel Master Switch. Horizontal stabilizer trimming and audio clicking shall resume.

(23) Depress copilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(24) Release copilot's Control Wheel Master Switch. Horizontal stabilizer movement and audio clicking shall resume.
(25) Release Secondary Pitch Trim Switch. Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(26) Perform steps (4) thru (25) with flaps extended to 8°. No audio clicking (cricket) shall occur.
(27) Perform steps (4) thru (25) with flaps extended to 20°. No audio clicking (cricket) shall occur.
(28) Retract flaps.
(29) Set Battery Switches off.
(30) Disconnect external power source from aircraft.

B. Trim-in-Motion System Operational Check *(Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior Aircraft modified per AAK 83-2.)*
(1) Connect external electrical power to aircraft.
(2) Set Battery Switches on.
(3) Ensure that flaps are fully retracted.
(4) Set Pitch Trim Selector Switch to PRI.
(5) Set and hold pilot's Trim and Trim Arming Switch to NOSE UP. Horizontal stabilizer leading edge shall lower and audio clicking (cricket) shall sound.
(6) Release pilot's Trim and Trim Arming Switch. Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(7) Set and hold pilot's Trim and Trim Arming Switch to NOSE DN. Horizontal stabilizer leading edge shall raise and audio clicking (cricket) shall sound.
(8) Depress pilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(9) Set and hold pilot's Trim and Trim Arming Switch to NOSE DN. Horizontal stabilizer leading edge shall raise and audio clicking (cricket) shall sound.
(10) Depress copilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(11) Repeat steps (5) thru (10) using copilot's Trim and Trim Arming Switch.
(12) Set Pitch Trim Selector Switch to OFF.
(13) Set pilot's Trim and Trim Arming Switch to NOSE UP. No horizontal stabilizer movement or audio clicking (cricket) shall occur.
(14) Set pilot's Trim and Trim Arming Switch to NOSE DN. No horizontal stabilizer movement or audio clicking (cricket) shall occur.
(15) Repeat steps (13) and (14) using copilot's Trim and Trim Arming Switch.
(16) Set Pitch Trim Selector Switch to SEC.
(17) Set Secondary Pitch Trim Switch to NOSE UP. Horizontal stabilizer leading edge shall lower and audio clicking (cricket) shall sound.
(18) Release Secondary Pitch Trim Switch. Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(19) Set and hold Secondary Pitch Trim Switch to NOSE DN. Horizontal stabilizer leading edge shall raise and audio clicking (cricket) shall sound.
(20) Depress pilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(21) Release pilot's Control Wheel Master Switch. Horizontal stabilizer trimming and audio clicking shall resume.
(22) Depress copilot's Control Wheel Master Switch (MSW). Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(23) Release copilot's Control Wheel Master Switch. Horizontal stabilizer movement and audio clicking shall resume.

(24) Release Secondary Pitch Trim Switch. Horizontal stabilizer movement shall cease and audio clicking (cricket) shall silence.
(25) Perform steps (4) thru (24) with flaps extended to 8°. No audio clicking (cricket) shall occur.
(26) Perform steps (4) thru (24) with flaps extended to 20°. No audio clicking (cricket) shall occur.
(27) Retract flaps.
(28) Set Battery Switches off.
(29) Disconnect external power source from aircraft.



Detail A

M35-274100-201-01

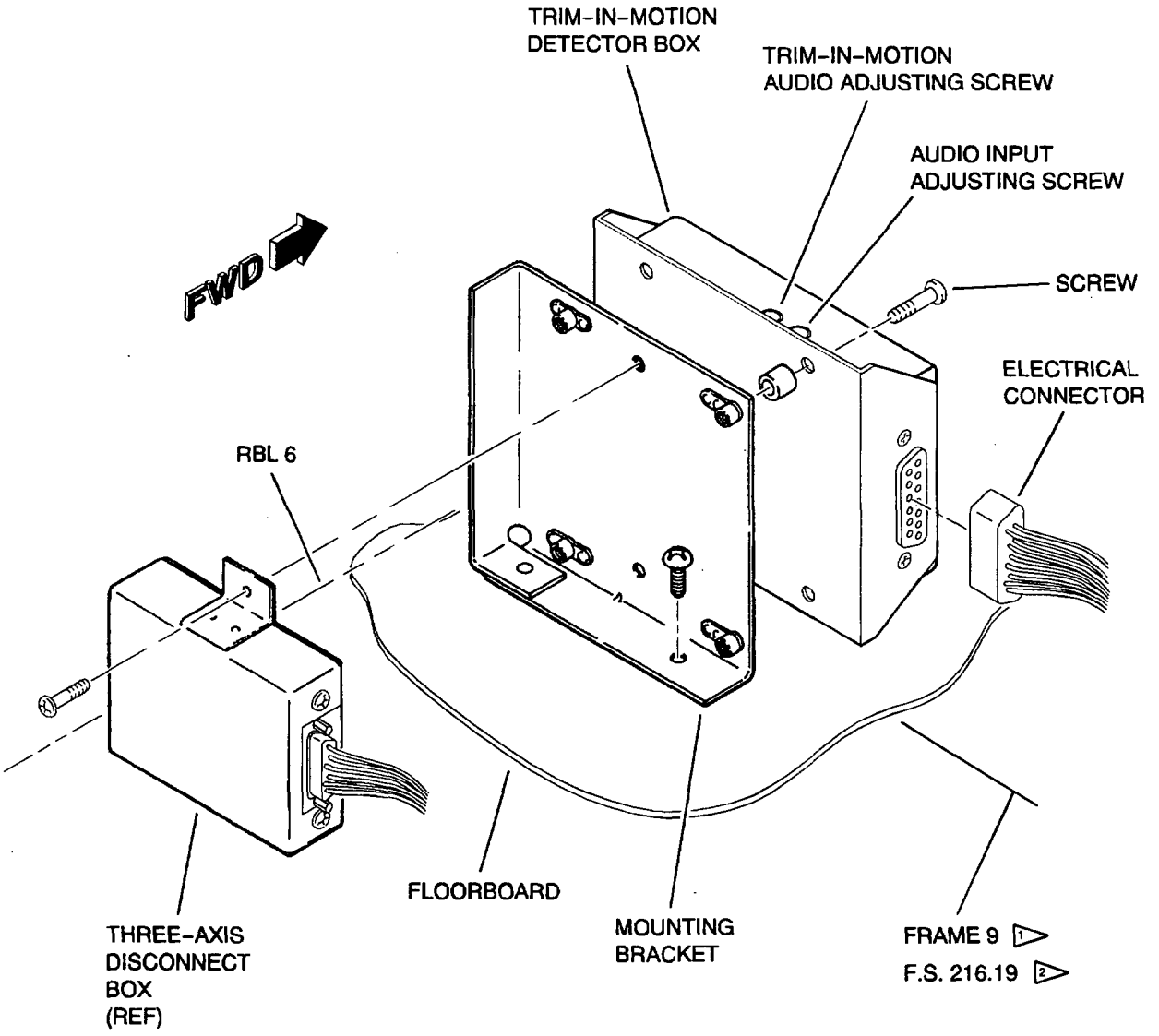
8-96B

Trim-In-Motion System Volume Adjustment
Figure 201 (Sheet 1 of 2)

EFFECTIVITY: AIRCRAFT 35-002 THRU 35-505, 36-002 THRU 36-053 NOT MODIFIED PER AAK 83-2

[1] AIRCRAFT 35-506 THRU 35-670;
36-054 THRU 36-063

[2] AIRCRAFT 35-671 AND SUBSEQUENT
36-064 AND SUBSEQUENT



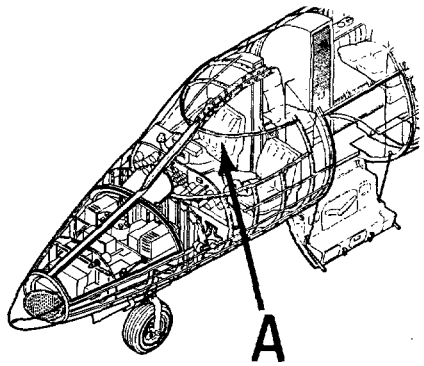
Detial A

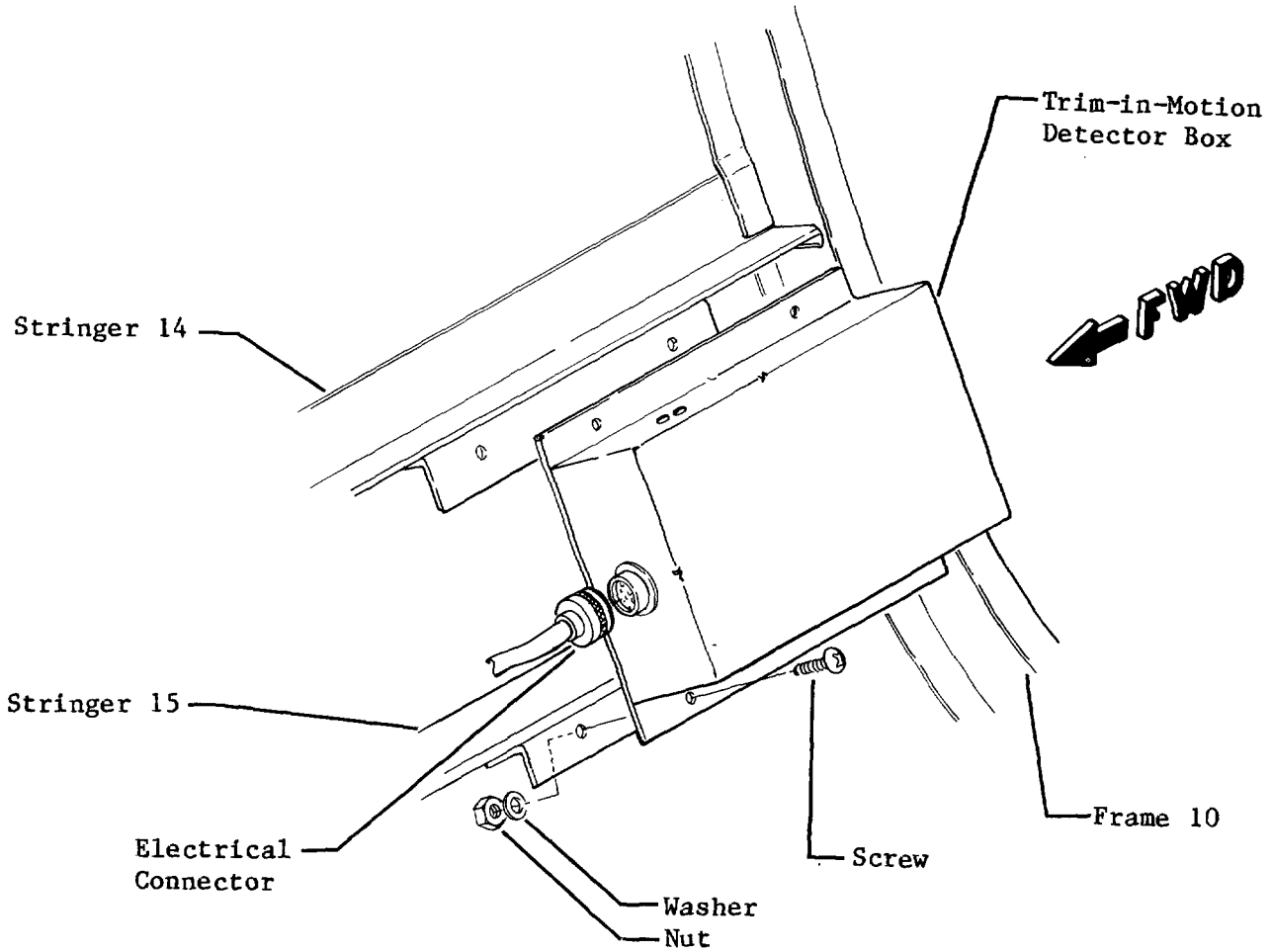
M35-274100-201-02

Trim-In-Motion System Volume Adjustment
Figure 201 (Sheet 2 of 2)

TRIM-IN-MOTION DETECTOR BOX - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Remove Trim-in-Motion Detector Box (*Aircraft 35-002 thru 35-505 and 36-002 thru 36-053 modified per AAK 83-2, "Installation of FC-530 Autopilot"*) (See figure 201.)
      (1) Remove electrical power from aircraft.
      (2) Gain access to trim-in-motion detector box behind upholstery on right-hand side of cockpit between stringers 14 and 15 forward of frame 10.
      (3) Disconnect electrical connector from detector box.
      (4) Remove attaching parts and detector box from aircraft.
   B. Install Trim-in-Motion Detector Box (*Aircraft 35-002 thru 35-505 and 36-002 thru 36-053 modified per AAK 83-2, "Installation of FC-530 Autopilot"*) (See figure 201.)
      (1) Position trim-in-motion detector box and secure with attaching parts.
      (2) Connect electrical connector to detector box.
      (3) Restore electrical power to aircraft.
      (4) Perform Trim-in-Motion System Volume Adjustment. (Refer to Adjustment/Test, 27-41-00.)
      (5) Install upholstery and equipment removed to gain access to detector box.
   C. Remove Trim-in-Motion Detector Box (*Aircraft 35-506 and Subsequent, 36-054 and Subsequent*) (See figure 201.)
      (1) Remove electrical power from aircraft.
      (2) Locate trim-in-motion detector box on floorboard behind copilot's seat.
      (3) Disconnect electrical connector from detector box.
      (4) Remove attaching parts and detector box from aircraft.
   D. Install Trim-in-Motion Detector Box (*Aircraft 35-506 and Subsequent, 36-054 and Subsequent*) (See figure 201.)
      (1) Position trim-in-motion detector box and secure with attaching parts.
      (2) Connect electrical connector to detector box.
      (3) Restore power to aircraft. Perform Trim-in-Motion System Operational Check. (Refer to Inspection/Check, 27-41-00.)
      (4) Perform Trim-in-Motion System Volume Adjustment. (Refer to Adjustment/Test, 27-41-00.)





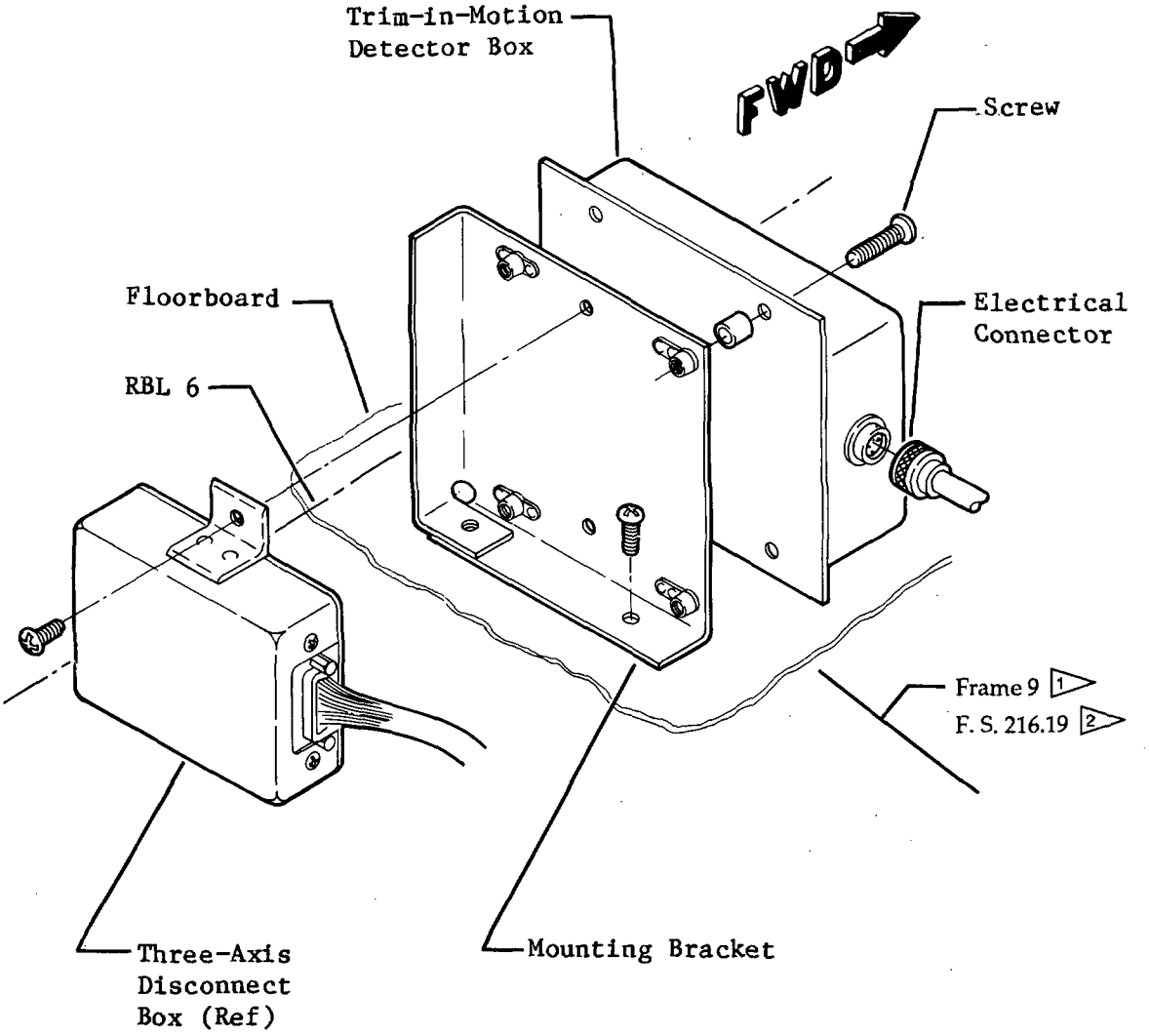
Detail A

8-96B

Trim-In-Motion Detector Box Installation
Figure 201 (Sheet 1 of 2)

1 *Aircraft 35-506 thru 35-670; 36-054 thru 36-063*

2 *Aircraft 35-671 and Subsequent 36-064 and Subsequent*



Detail A

8-96B-2

Trim-In-Motion Detector Box Installation
Figure 201 (Sheet 2 of 2)

FLAP SYSTEM - DESCRIPTION AND OPERATION

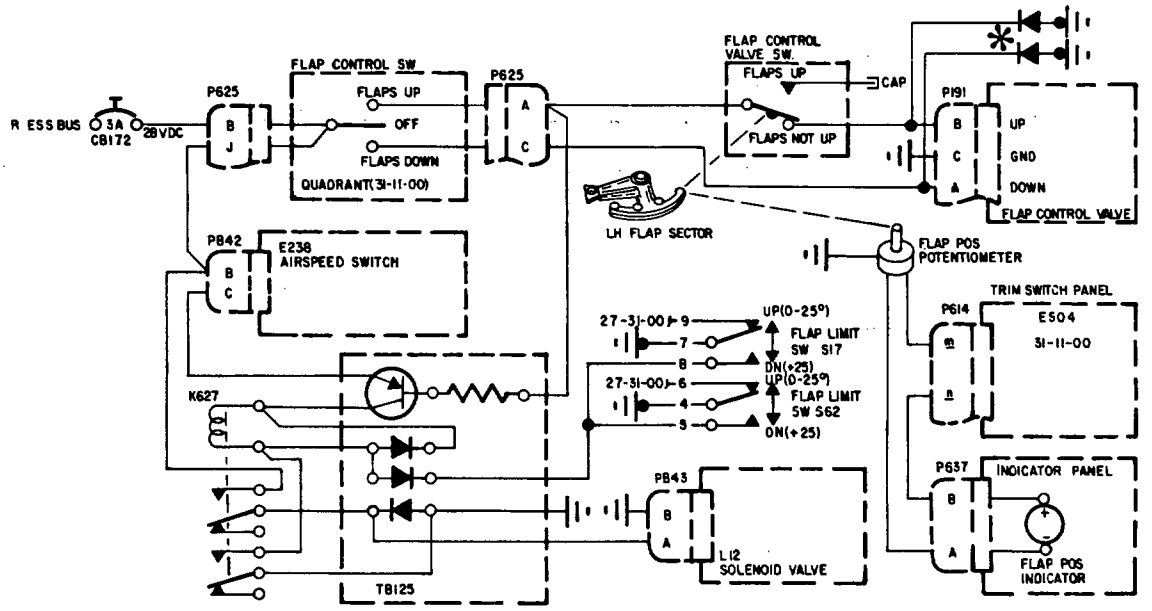
1. **Description**
   A. The flap system is electrically controlled and hydraulically operated and provides lift to the wing when partially extended and increases drag to reduce speed when fully extended.
   B. *Aircraft 35-417, 35-419, 35-477, 35-479, 35-483 and Subsequent and 36-051 and Subsequent* are equipped with a gated flap control system. (Refer to 27-51-00 for coverage of the gated flap control system.)
   C. The single-slotted flaps are attached to the inboard rear wing spar by tracks, rollers and hinges. The dual actuator flap system is electrically controlled and hydraulically operated. A flap switch is located on the cockpit center pedestal and a flap position indicator is located on the center instrument panel. An aural warning system is electrically connected to alert the pilot that the flaps have been fully extended without the landing gear being extended.
   D. On *Aircraft 35-002 thru 35-059 and 36-002 thru 36-017*, the flap system consists of two flaps, two flap actuators, a control valve, a check valve, a restrictor, a flap blowup relief valve, a flap load-limiting relief valve, an airspeed switch and a solenoid-actuated load-limiting valve.
   E. On *Aircraft 35-060 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482, and 36-018 thru 36-050*, the flap system consists of two flaps, two flap actuators, a control valve, a check valve, a restrictor, and a flap blowup relief valve.
   F. Maximum flap extension is 40° (+5°/-0°) down. Flaps should travel from maximum up to maximum down in 6 seconds or less.
   G. When the flaps are extended below 25°, the spoilers (in the augmentation mode) work independently to assist the ailerons. During the aileron augmentation mode, the spoilers are controlled by an electrical aileron monitoring system. (Refer to 27-60-00.)

2. **Operation** (See figures 1 thru 3.)
   A. Setting the Flap Switch on the center pedestal to the DN position energizes a solenoid which positions the flap control valve to direct hydraulic fluid to the down port of the flaps actuators. The actuators rotate sectors connected to the flap push-pull tube and extends flaps. Interconnecting control cables attached to flap sectors ensure synchronized flap travel.
   B. When the Flap Switch is returned to the off position, the solenoid is de-energized and the flap control valve returns to neutral. This cuts off all hydraulic flow to the actuators and stops the flaps at the desired position.
   C. Setting the Flap Switch to UP energizes the solenoid and positions the flap control valve selector to direct hydraulic fluid to the up port of the flap actuators. This rotates the flap sectors which retract the flaps. On *Aircraft 35-002 thru 35-531 and 36-002 thru 36-053 not modified per SB35/36-27-15*, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference," when the flaps are fully retracted, the flap control valve switch (S16) is contacted. This breaks the electrical circuit to de-energize the solenoid and allows the flap control valve to return to the neutral position and stop hydraulic flow.
   D. An aural warning system is electrically connected to the flaps and landing gear. With the flaps extended beyond 25°, an aural warning horn will sound until the landing gear is fully extended and locked.
   E. After extended parking time with power shut off, the flaps may droop and spoilers rise. This is normal. On starting either the engines or the auxiliary hydraulic pump, the flaps and spoilers will return to the position prior to shutoff provided the control switch settings have not been changed.
   F. A check valve, installed in the pressure inlet to the flap control valve, will prevent the flaps from retracting in the event of hydraulic system failure upstream of the flap control valve when the flaps are extended.
   G. A 1650 psi pressure relief valve, installed in the flap down line, relieves pressure to prevent excessive structural loads.

H. On *Aircraft 35-002 thru 35-059 and 36-002 thru 36-017*, a solenoid-operated flap load-limiting valve is installed in the flap down line. The solenoid is electrically controlled by an airspeed switch when the aircraft speed exceeds the aircraft flap placard speed with the flaps extended below 25°. The solenoid, when energized allows the load-limiting valve to block the flap pressure line through a 1000 psi load-limiting relief valve.

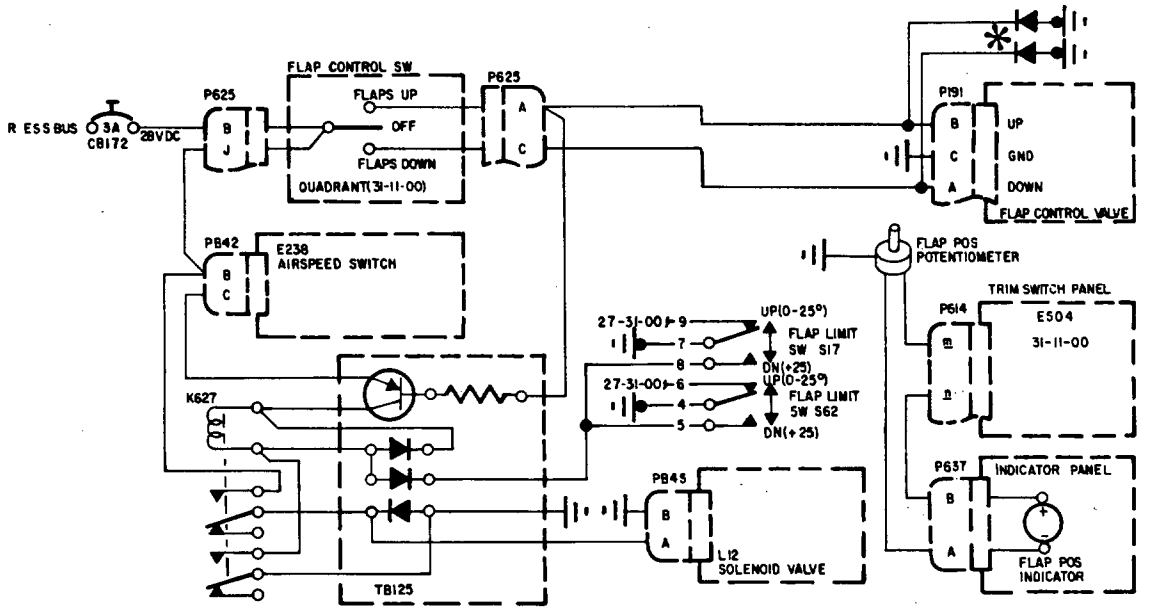
I. On *Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not equipped with AAK 76-4 (Reduced Approach Speed System)*, there are four cam-operated position switches adjacent to the sectors. The 25° position switch at the RH flap sector provides flap position information to the stall warning bias box, gear aural warning, and aileron augmentation systems. The 25° position switch at the LH flap sector provides flap position information to the stall warning bias box and (on *35-002 thru 35-059 and 36-002 thru 35-017)* the flap load-limiting system. Both 13° position switches, one located at each flap sector, provide flap position information to the stall warning bias box and spoiler warning system.

J. On *Aircraft 35-067 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482, 36-018 thru 36-050, and earlier Aircraft equipped with AAK 76-4 (Reduced Approach Speed System)*, there are two, three-layer, rotary-type position switches, one connected to each flap sector. Each switch provides 3°, 13°, and 25° flap position signals. All three layers of each switch provide flap position information to the stall warning computer. The 25° and 13° layers of both switches provide flap position information to the gear aural warning system. The 13° layer of each switch also provides flap position information to the autopilot electric box.



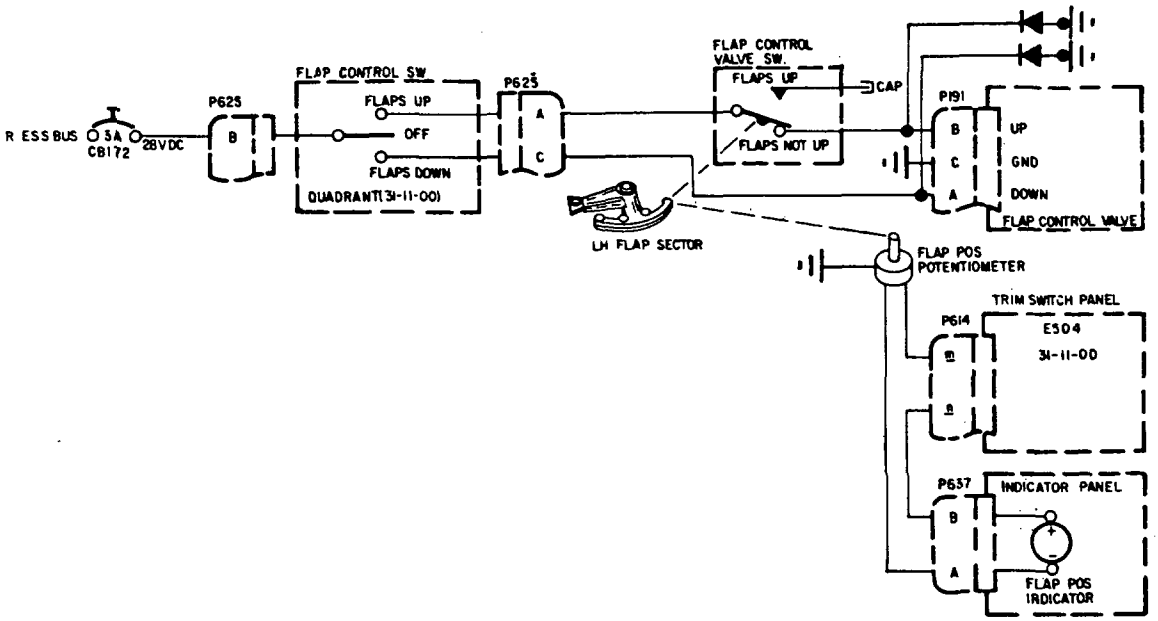
Aircraft 35-002 thru 35-059 and 36-002 thru 36-017, not modified per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference."

* *ON AIRCRAFT 36-002,* DIODE NOT EFFECTIVE

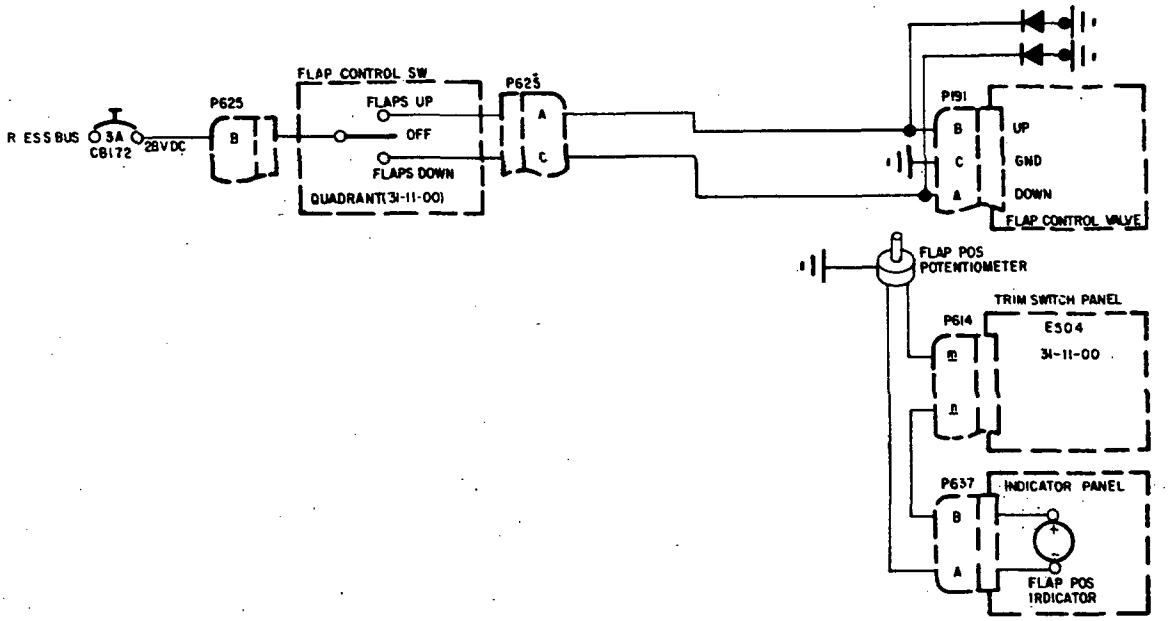


Aircraft 35-002 thru 35-059 and 36-002 thru 36-017, modified per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference."

Flap Electrical Control Schematic
Figure 1 (Sheet 1 of 2)

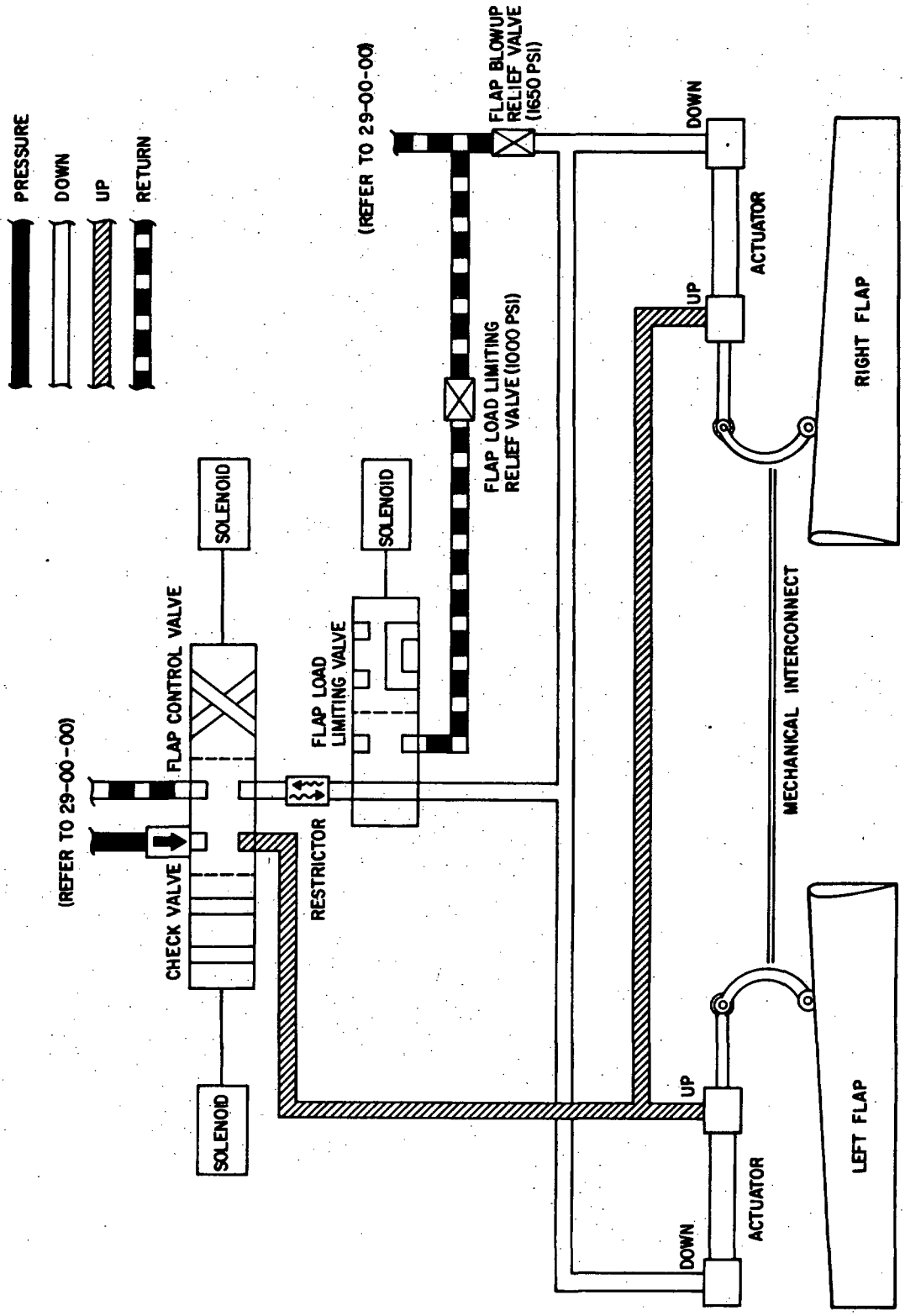


Aircraft 35-060 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482 and 36-018 thru 36-050, not modified per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference."

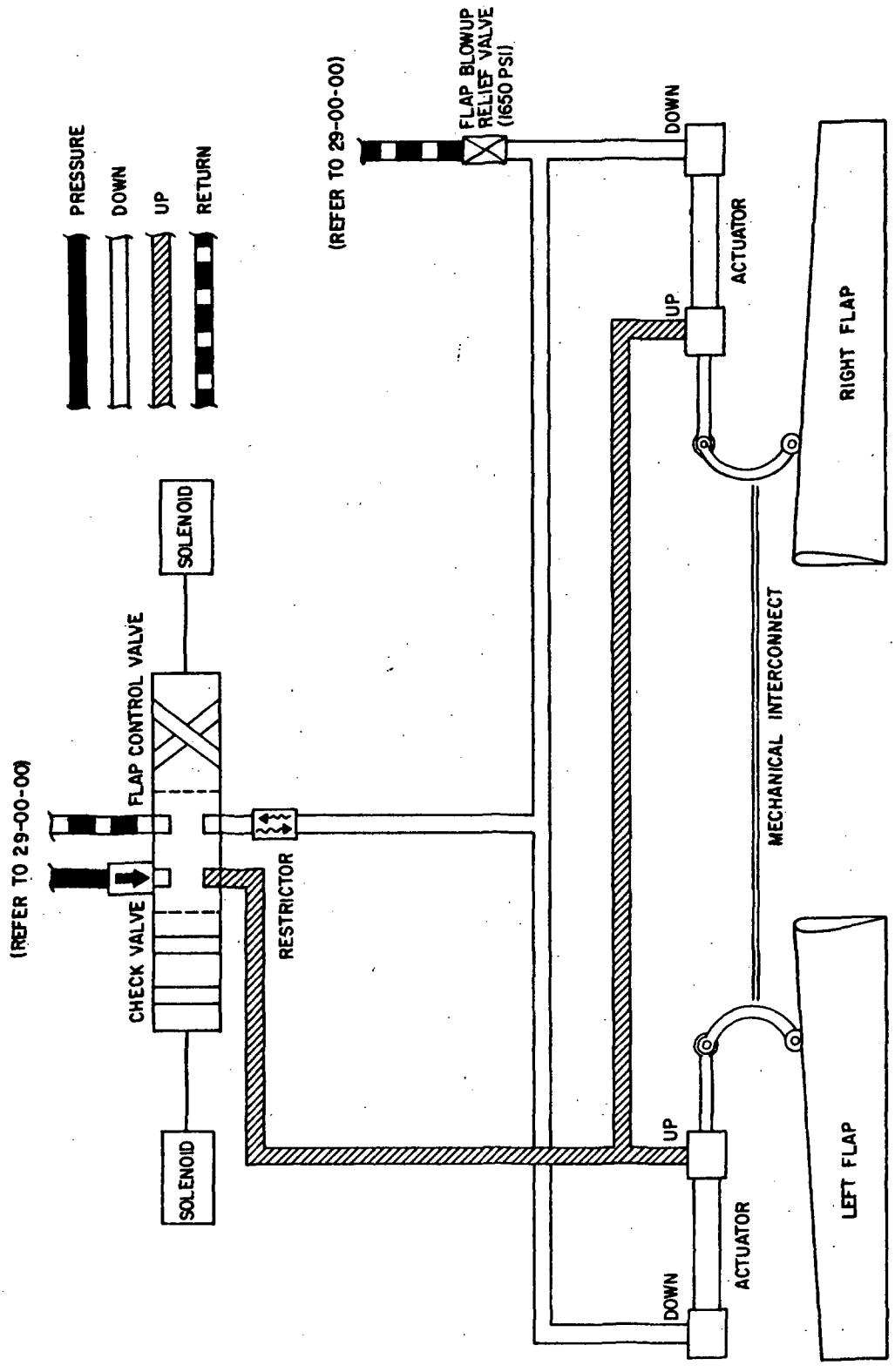


Aircraft 35-060 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482 and 36-018 thru 36-050, modified per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference."

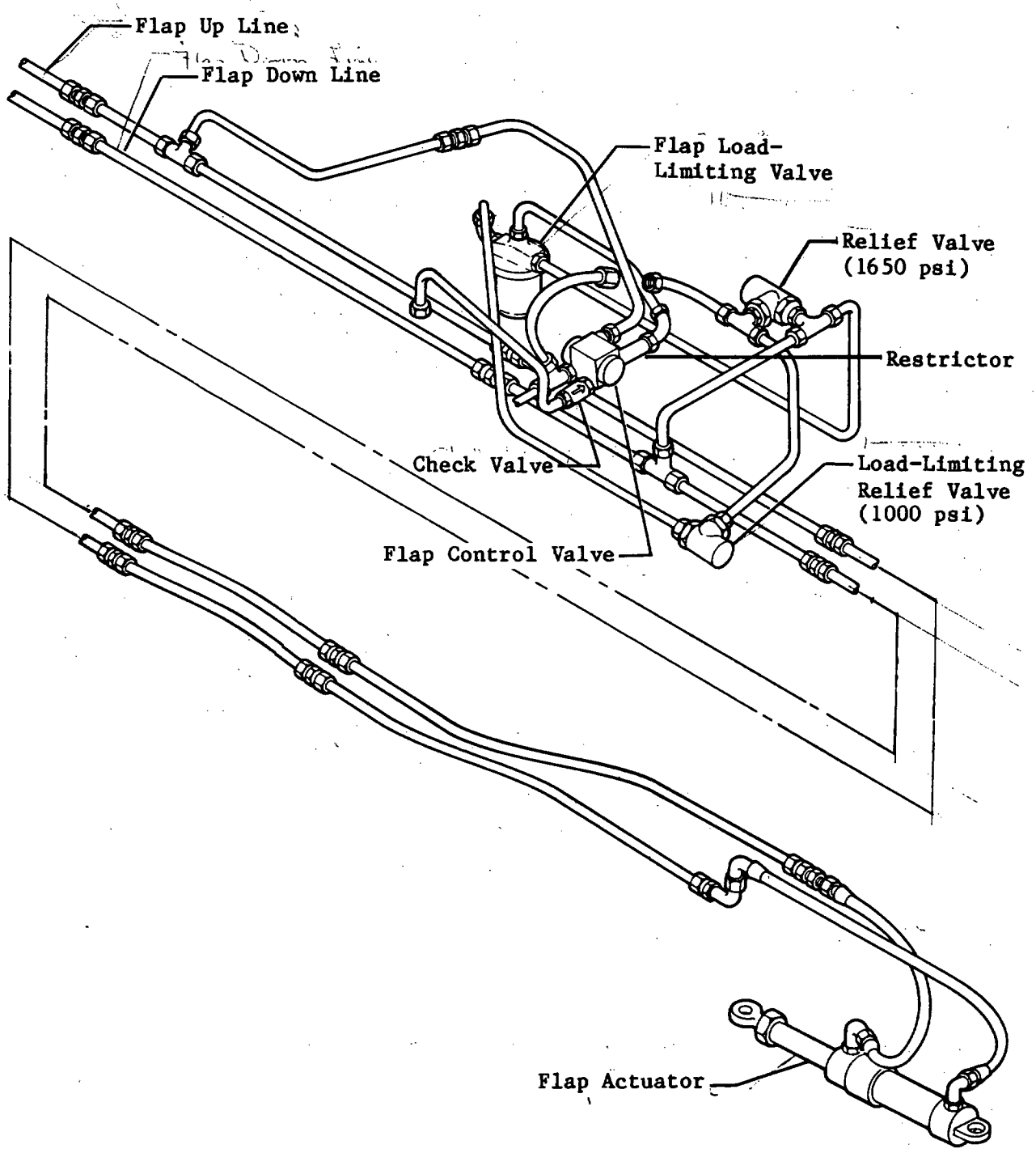
Flap Electrical Control Schematic
Figure 1 (Sheet 2 of 2)



Flap System Hydraulic Schematic
Figure 2 (Sheet 1 of 2)

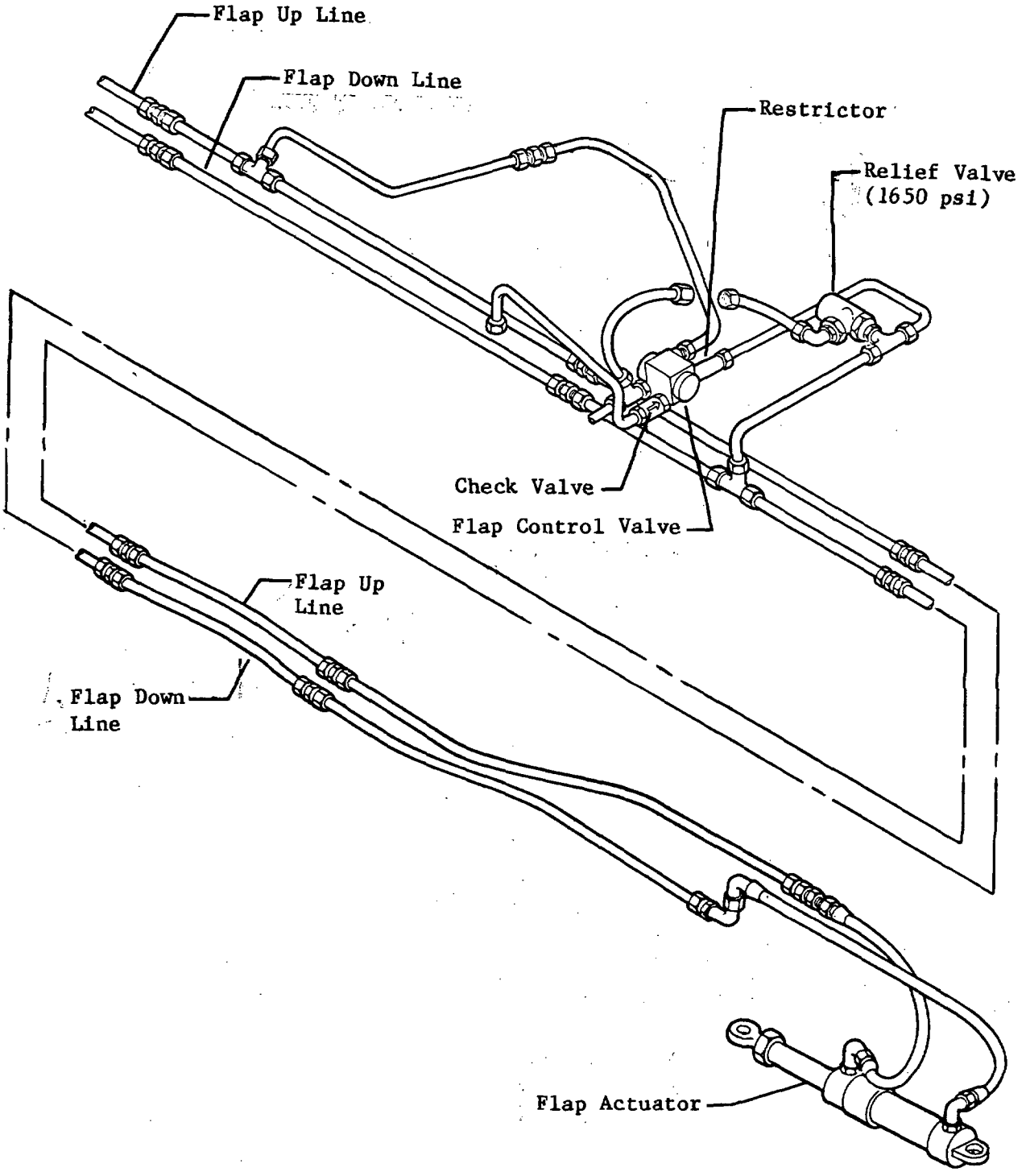


Flap System Hydraulic Schematic
Figure 2 (Sheet 2 of 2)



Flap Hydraulic System Components Installation
Figure 3 (Sheet 1 of 2)

EFFECTIVITY: 35-002 THRU 35-059, AND 36-002 THRU 36-017



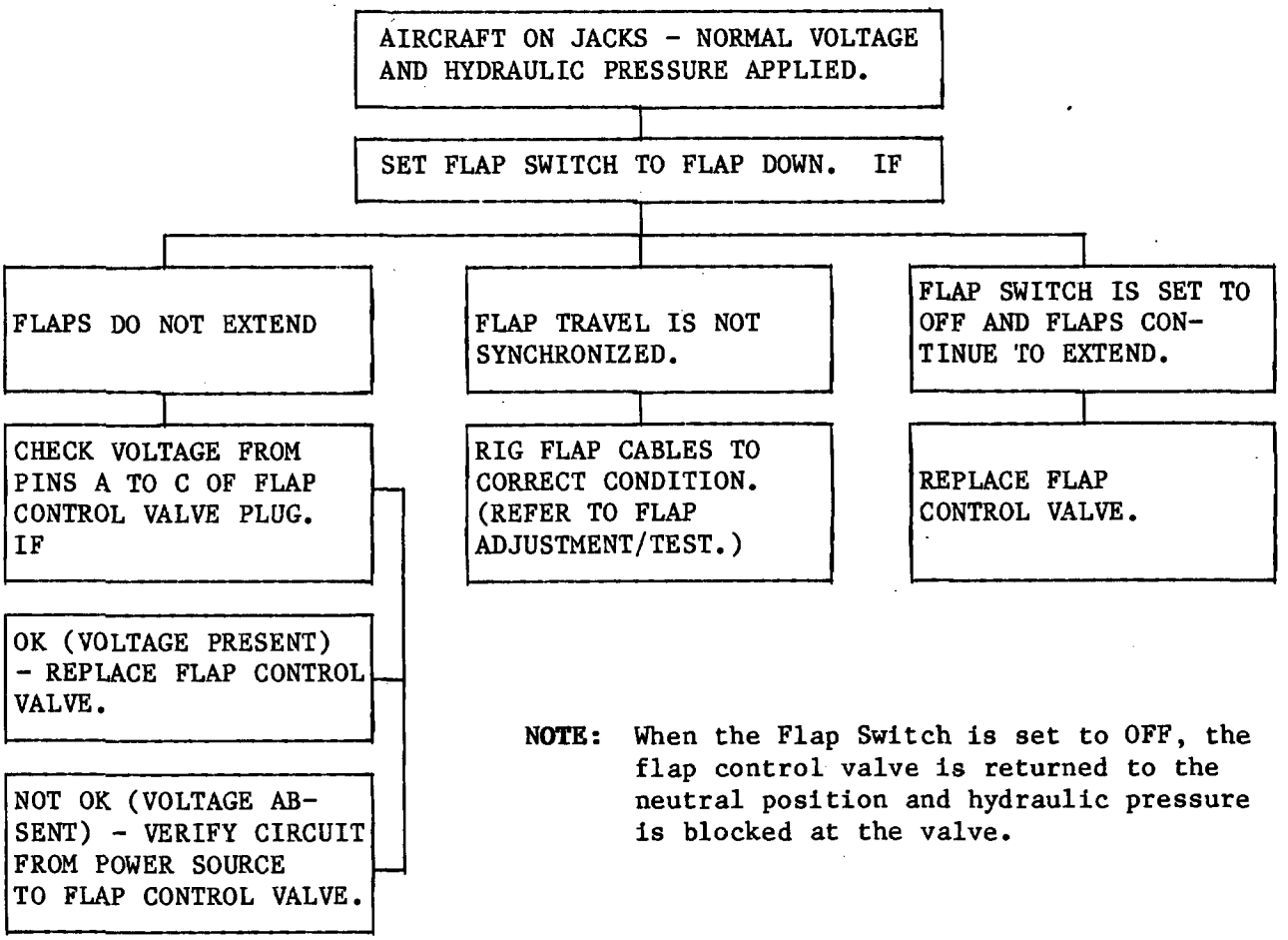
Flap Hydraulic System Components Installation
Figure 3 (Sheet 2 of 2)

FLAP SYSTEM - TROUBLE SHOOTING

1. TROUBLE SHOOTING FLAP SYSTEM

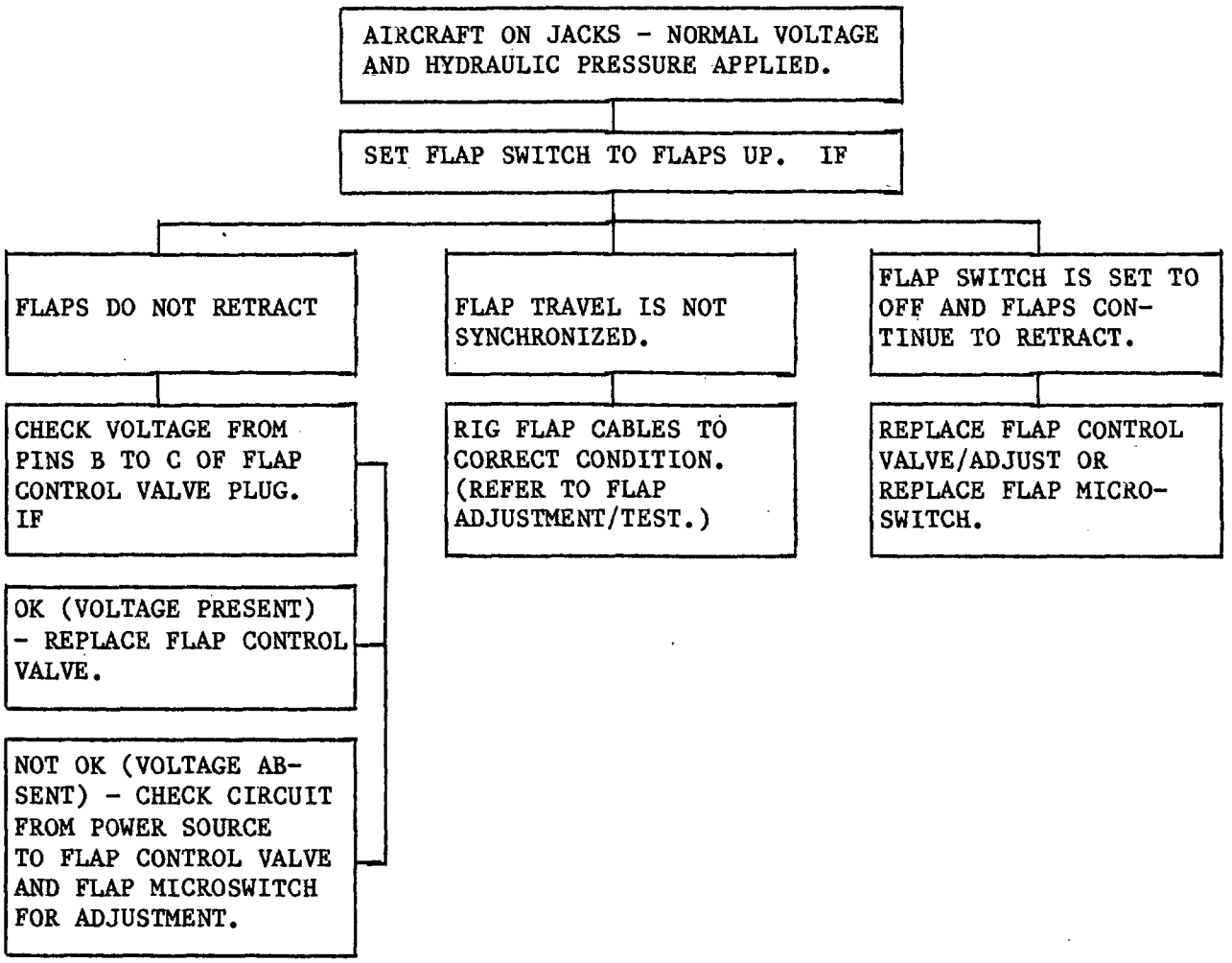
NOTE: The aircraft must be placed on jacks and normal system voltage applied to properly trouble shoot the flap system.

A. See figure 101 for a guide to trouble shooting the flap system.



NOTE: When the Flap Switch is set to OFF, the flap control valve is returned to the neutral position and hydraulic pressure is blocked at the valve.

Trouble Shooting Chart - Flap System
Figure 101 (Sheet 1 of 2)



Trouble Shooting Chart - Flap System
Figure 101 (Sheet 2 of 2)

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Tensiometer		Commercially Available	Adjust cable tension.
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	Measure angles.
Digital Protractor (DP-45) (Alternate)	02475-01	Lucas & Schaevitz Pennsauken, NJ	Measure angles.
Rigging Pin		Local Manufacture	Cable rigging.
Pitot-Static Tester	1811F	Barfield Instrument Co. Atlanta, GA	Test system.

2. Adjustment/Test

NOTE:
- Flap system rigging consists of synchronizing the flaps, adjusting cable tension and adjusting the flap limit switch.
- Rigging pin is fabricated from 3/16 inch round steel stock. Make 90° bend, approximately 8-1/2 inches from end of 12 inch piece of stock. Paint short end of piece red (for visibility).

A. Rig Flap System (See Figure 201.)

WARNING: REMAIN CLEAR OF SPOILERS DURING OPERATION TO PREVENT BODILY INJURY.

NOTE: The flaps shall be synchronized to within 2° of each other at all times.

(1) Remove flap sector access cover.
(2) Adjust flap control cable tension in accordance with procedures outlined in step 2.D.
(3) With flaps in retract position, align rigging pin holes in sector with holes in bracket, and insert rigging pins.
(4) Disconnect actuator rod ends from both left and right sectors.
(5) Check for 0.03 inch minimum clearance between flap sectors and wing spar 8.
(6) Adjust flap push-pull tubes until flap rollers contact forward end of flap tracks.
(7) With hydraulic actuators fully retracted (bottomed internally), adjust rod end terminals so that terminals are aligned with holes in sectors where actuators are normally connected.
(8) Turn rod ends 1/2 turn clockwise (this shortens retracted length of the cylinders approximately 0.02 inch), to preload flap track rollers snug into forward end of flap nose track.
(9) Attach rod end terminals to sectors and safety wire terminals.
(10) On *Aircraft not modified per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference"*, adjust flap control valve microswitch to actuate.

(11) Remove rigging pins.
(12) Using the Flap Switch on the Center Pedestal, check for proper flap travel. Full down travel shall be 40 (+5, -0)°.
(13) With flaps in extended position, spanwise movement (end play) shall be 0.34 inch maximum. If spanwise movement exceeds limits, replace buttons on cam followers. Ensure that there is a positive clearance between nose roller and aft end of nose roller track slot (4 places).
(14) Retract flaps to the 20° down position. With all flap end play positioned to the extreme outboard position, clearance between the flap and aileron and the flap and aileron trim tab shall be a minimum of 0.100 inch.
(15) Inspect to ensure that the following items do not occur:
(a) Interference between flap sector and switch mounting bracket.
(b) Contact between flap and spoiler actuator bracket.
(c) Contact between flap and switch actuating arm on spoiler.
(d) Contact between flap and lower wing surface. Make any necessary adjustments.
(16) Install access covers.
(17) Perform operational checks of Stall Warning System (27-31-00), Spoiler Warning System (27-60-00), and the Landing Gear Aural Warning System (Chapter 32); make adjustments as required.

B. Functional Test of Flap System
(1) Lower tailcone access door.
(2) Connect a hydraulic power unit to pressure and return quick-disconnects.
(3) Set Battery Switches to BAT 1 and BAT 2.
(4) Extend flaps. During flap extension, observe flap position indicator. Stop flaps at each marked interval until flaps are full down.
(5) Retract flaps. During flap retraction, observe that indicator pointer movement from the DN to the UP position is smooth and uninterrupted. Set Battery Switches to OFF.
(6) Disconnect hydraulic power unit and install caps on hydraulic pressure and return quick-disconnects.
(7) Secure tailcone access door.

C. Functional Test of Flap Load-Limiting System (*Aircraft 35-002 thru 35-059 and 36-002 thru 36-017*)
(1) Connect a pitot-static tester to the copilot's pitot-static system.
(2) Extend the flaps to the full down position.

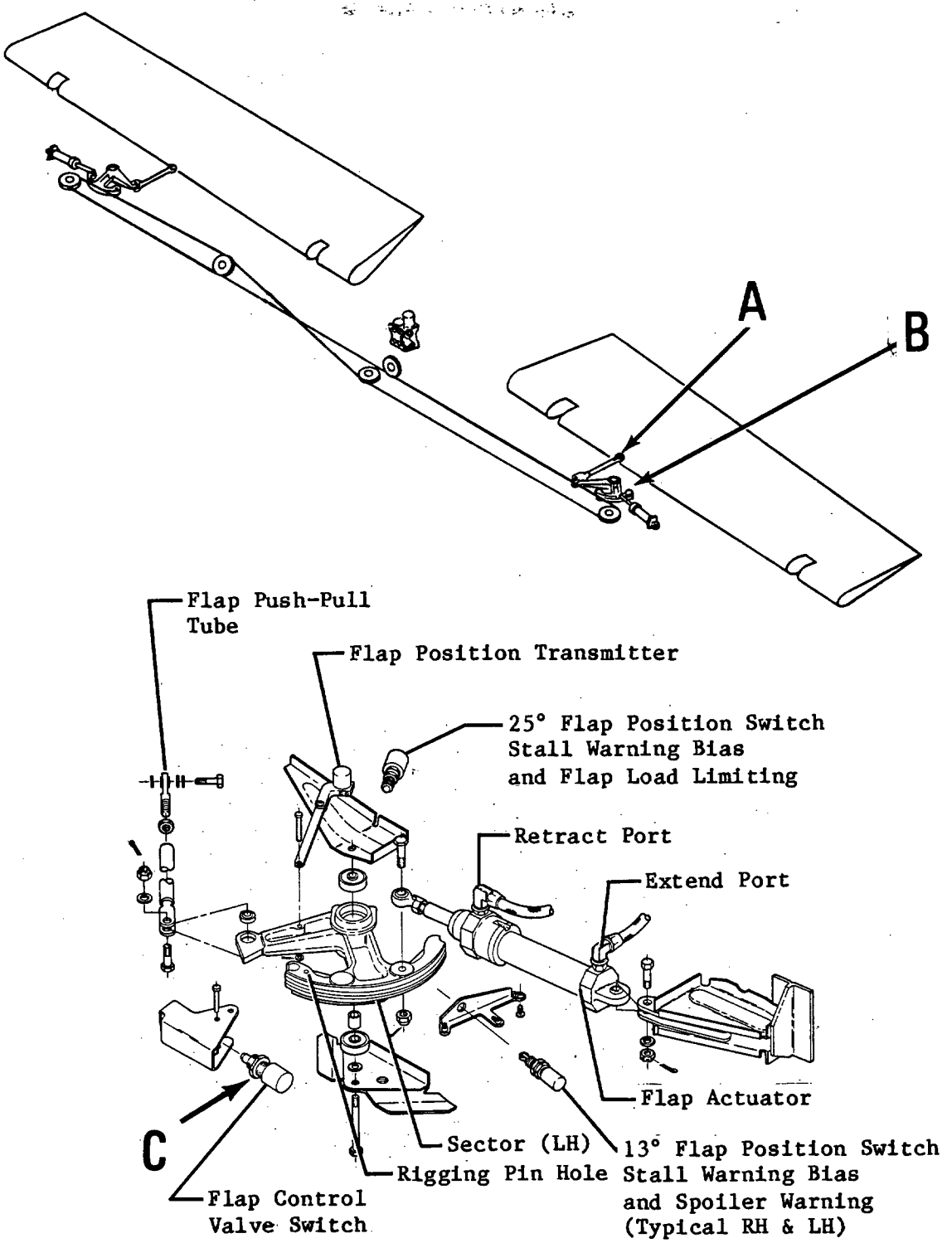
CAUTION: APPLY ONLY PRESSURE (NOT VACUUM) TO THE PITOT LINES. APPLY PRESSURE VERY SLOWLY UNTIL 80 KNOTS IS REACHED; PRESSURE APPLICATION AFTER THIS POINT MUST NOT EXCEED 20 KNOTS PER SECOND OR INSTRUMENT DAMAGE COULD RESULT.

NOTE: Station a person in the area of the flap load limiting valve to listen for the valve to actuate.

(3) Slowly pressurize the pitot system until the airspeed indicator reads 160 knots. Verify that the flap load-limiting valve is energized (click) when pressurized at 160 knots.

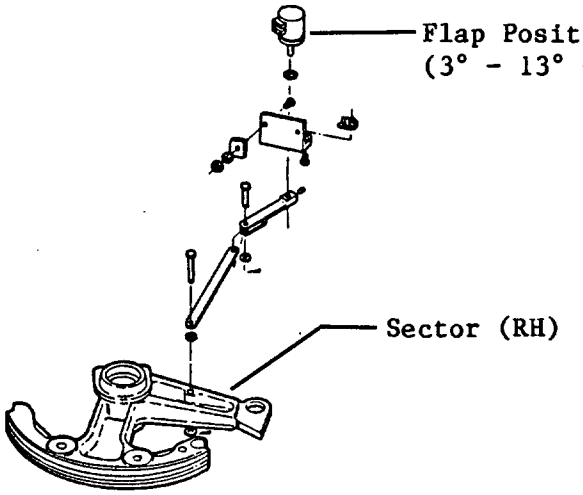
CAUTION: DEPRESSURIZATION RATE SHOULD NOT EXCEED 20 KNOTS PER SECOND OR INSTRUMENT DAMAGE COULD RESULT.

(4) Slowly decrease pressure in the pitot system. Verify that the flap load-limiting valve de-energizes (click) when the airspeed indicator reads 151 knots.
(5) Retract the flaps to the full up position.
(6) Slowly pressurize the pitot system until the airspeed indicator reads 160 knots. Verify that the flap load-limiting valve does not energize with the flaps full up.



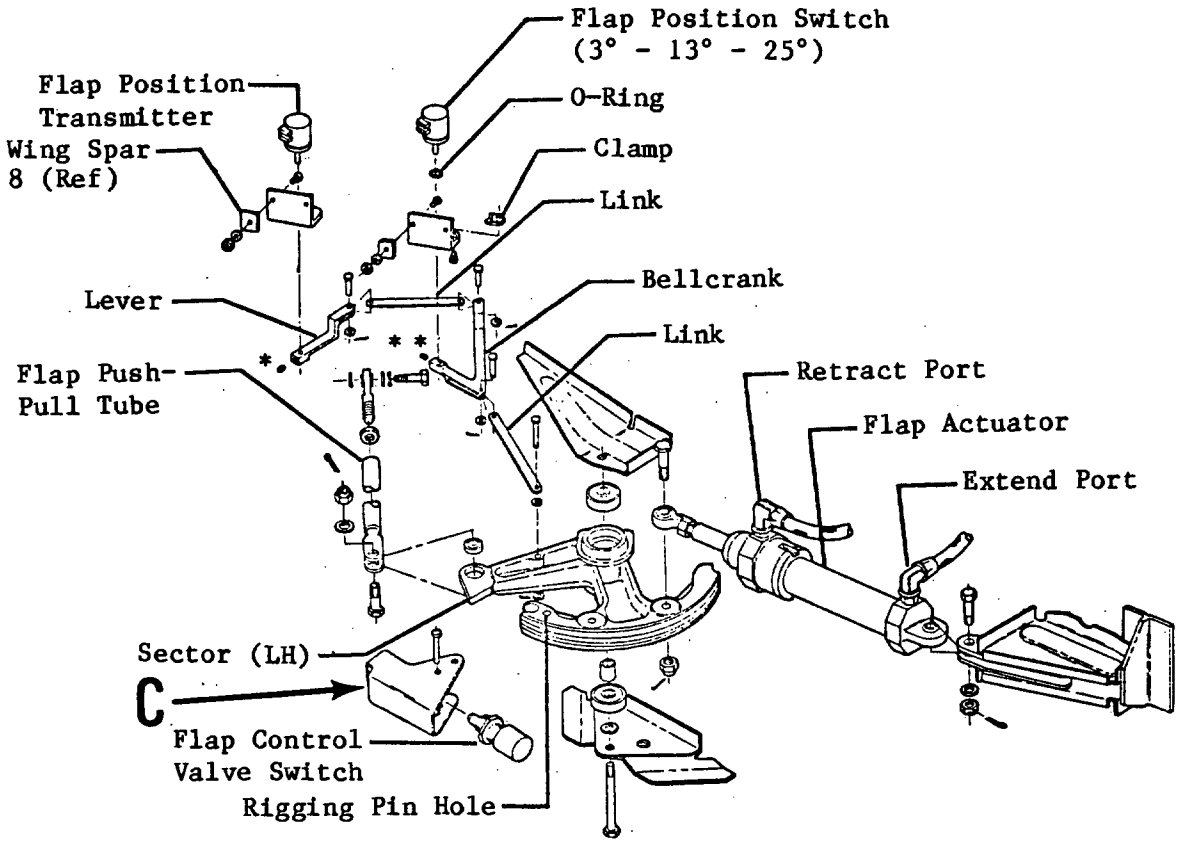
Detail B

Flap System Rigging
Figure 201 (Sheet 1 of 3)



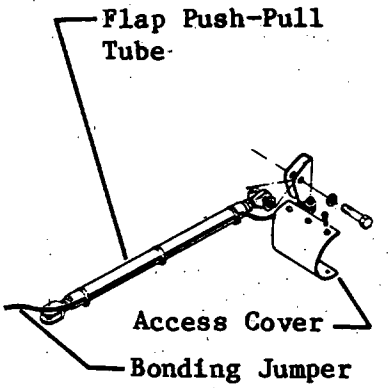
**Screw, washer, and nut installed on Aircraft 35-446 thru 35-476, 35-478, 35-480 thru 35-482.

*Screw, washer, and nut installed on Aircraft 35-410 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482, 36-045 thru 36-050.

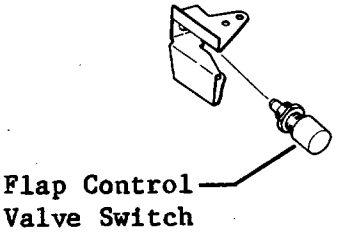


Detail B

Flap System Rigging
Figure 201 (Sheet 2 of 3)



Detail A



Applicable Only To Aircraft Not Modified Per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets For Cracks And Flap System For Interference"

Detail C

Flap System Rigging
Figure 201 (Sheet 3 of 3)

(7) With the pitot system pressurized to 160 knots, lower the flaps below 25° but not full down. Retract flaps and verify that the flaps can be retracted.

CAUTION: DEPRESSURIZATION RATE SHOULD NOT EXCEED 20 KNOTS PER SECOND OR INSTRUMENT DAMAGE COULD RESULT.

(8) Depressurize pitot system.
(9) Disconnect pitot-static tester.

D. Adjust Flap Control Cable Tension

NOTE:
- Prior to flap cable tension adjustment, torque wrench tensiometer adapter calibration must be accomplished. (Refer to Adjustment/Test, 27-00-00.)
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The Cable Tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to Adjustment/Test, 27-00-00.)

(1) With flaps in retract position, remove flap sector access cover, align rigging pin holes in sector with holes in bracket, and insert 3/16" rigging pin.
(2) Remove wing access panel, located on spar 7 in the left wheel well, to gain access to flap control cables and turnbuckles.
(3) Remove locking clips from turnbuckles.
(4) If a new cable is being installed, perform the following:
  (a) Adjust flap cable tension to 200 (±10) pounds (90.7 [±4.5] kg).
  (b) Cycle flaps 25 times to seat in and pre-stretch new cable.
  (c) Reduce cable tension.
(5) Adjust flap control cable tension to 100 (±25) pounds (45.4 [±11.3] kg).

NOTE:
- The threads at opposite ends of the turnbuckle barrel shall be started at the same time so thread engagement on both ends will be approximately equal.
- The final adjusted turnbuckle combination shall have no more than three threads of the terminal, fork, or eye exposed outside the barrel of the turnbuckle.

(6) Once proper cable tension has been obtained, remove rigging pin and operationally check flap control system. (Refer to Inspection/Check, this section.)
(7) Install locking clips in turnbuckles.
(8) Install all previously removed access panels.
(9) Restore aircraft to normal.

3. Inspection/Check

A. Perform Flap Control Cable Tension Check (See Figure 202.)

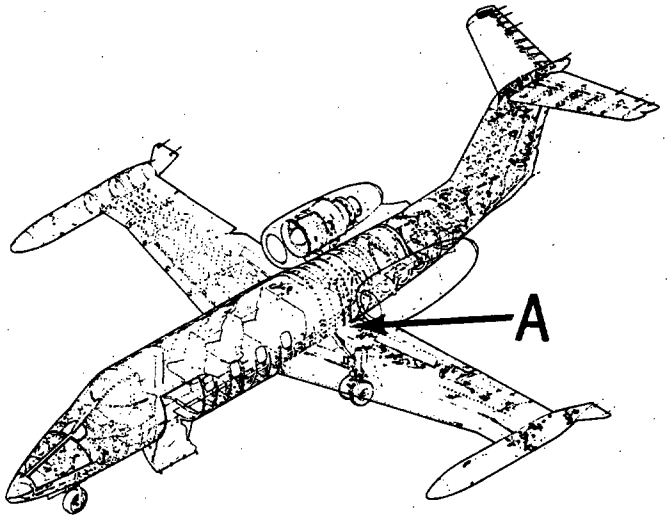
NOTE:
- Perform flap control cable tension check in accordance with the current inspection intervals specified in Chapter 5.
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The cable tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to 27-00-00, Adjustment/Test.)

(1) Remove wing access panel, located on spar 7 in the left wheel well, to gain access to flap control cables and turnbuckles.

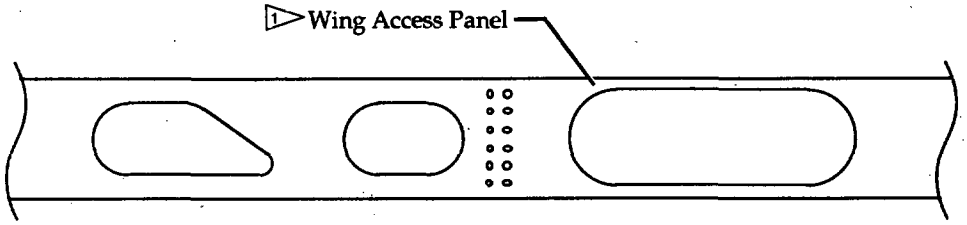
(2) Attach tensiometer to flap control cables. Flap cable tension shall be 100 (±25) pounds at 75°F (45.3 [±11.4] kg at 24°C).

NOTE:
- The flap cable must be lying against the back of the tensiometer when pressure is applied so that no extra pull is applied to the cable.
- Avoid pushing or pulling on tensiometer or cable while a reading is being taken.

(3) Adjust flap control cable tension if readings do not fall within tolerances. (Refer to Adjustment/Check, this section.)
(4) Remove tensiometer from flap control cable and install wing access panel.
(5) Restore aircraft to normal.



1 Flap control cables and turnbuckles located behind wing access panel.



(VIEW LOOKING AFT AT SPAR 7, LH WHEEL WELL)

Detail A

Flap Control Cable Tension Check
Figure 202

1. Removal/Installation

NOTE:
- Removal and installation of both flaps is typical.
- When installing a new flap it is necessary to trim the flap fairing to proper fit. Two installation procedures are provided, one for reinstalling the existing flap and one for installing a new flap.

A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

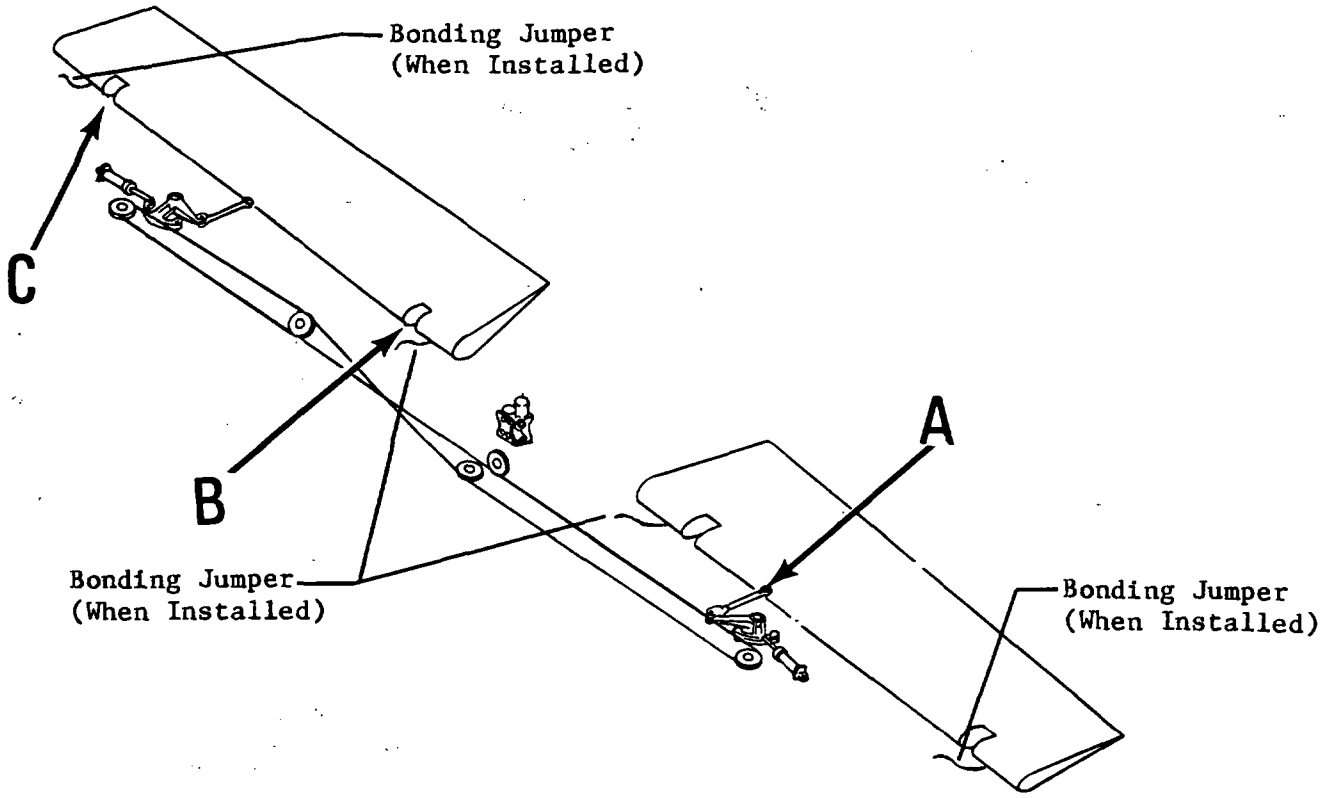
NAME	PART NUMBER	MANUFACTURER	USE
Grease	MIL-G-81322	Commercially Available	Lubrication.
Zerk Gun		Commercially Available	Apply grease.
90° Adapter	327265	Alemite Co.	Apply grease.
Clean, Cotton Cloth		Commercially Available	Cleaning.
Protective Tape	#8681	3M Company St. Paul, MN	Prevent wing/ flap chafe condition.
Protective Tape	#8641	3M Company St. Paul, MN	Prevent wing/ flap chafe condition.
Isopropyl Alcohol		Commercially Available	Clean surface.
Adhesion Promoter	#86	3M Company St. Paul, MN	Prepare surface for tape.

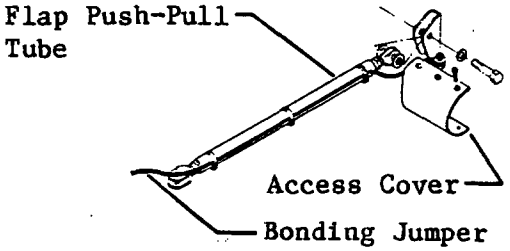
B. Remove Flap (See Figure 201.)

WARNING: **SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM THE AIRCRAFT WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELECTRICAL POWER.**

(1) With hydraulic and electrical power supplied to aircraft, fully extend spoilers and flaps.
(2) Remove hydraulic power from aircraft.
(3) Remove electrical power from aircraft.
(4) Remove flap sector access cover from underside of wing and insert rigging pin in sector.

NOTE: Rigging pin is fabricated from 3/16 inch round steel stock. Make 90° bend, approximately 8-1/2 inches from end of 12 inch piece of stock. Paint short end of piece red (for visibility).

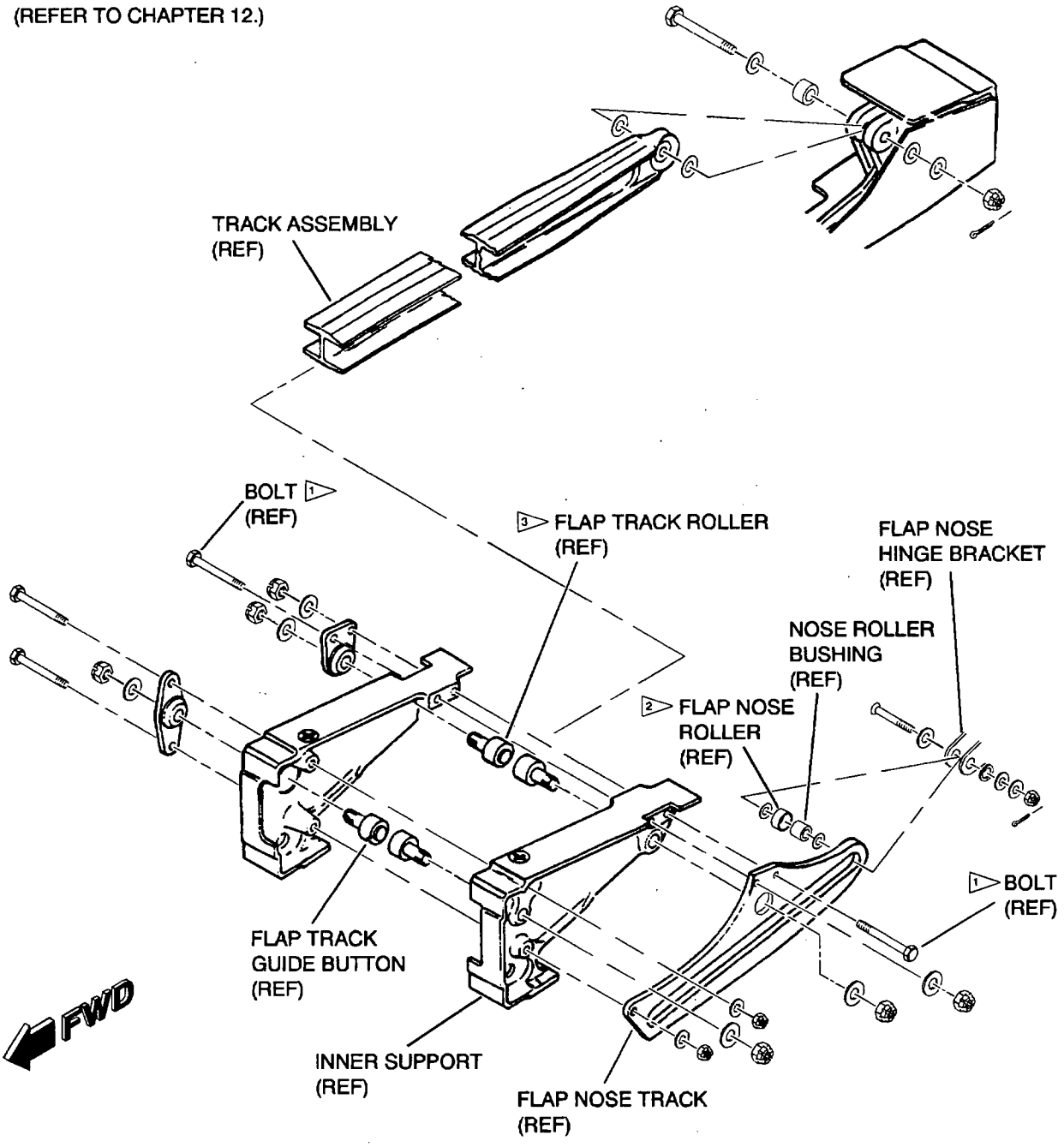




Detail A

Flap Installation
Figure 201 (Sheet 1 of 2)

1 BE SURE HEADS OF THESE BOLTS ARE LOCATED AS SHOWN. IF NOT INSTALLED AS SHOWN, BOLTS WILL NOT CLEAR WHEN FLAPS ARE RETRACTED.
2 DO NOT LUBRICATE THE FLAP NOSE ROLLERS.
3 LUBRICATE FLAP TRACK ROLLERS. (REFER TO CHAPTER 12.)

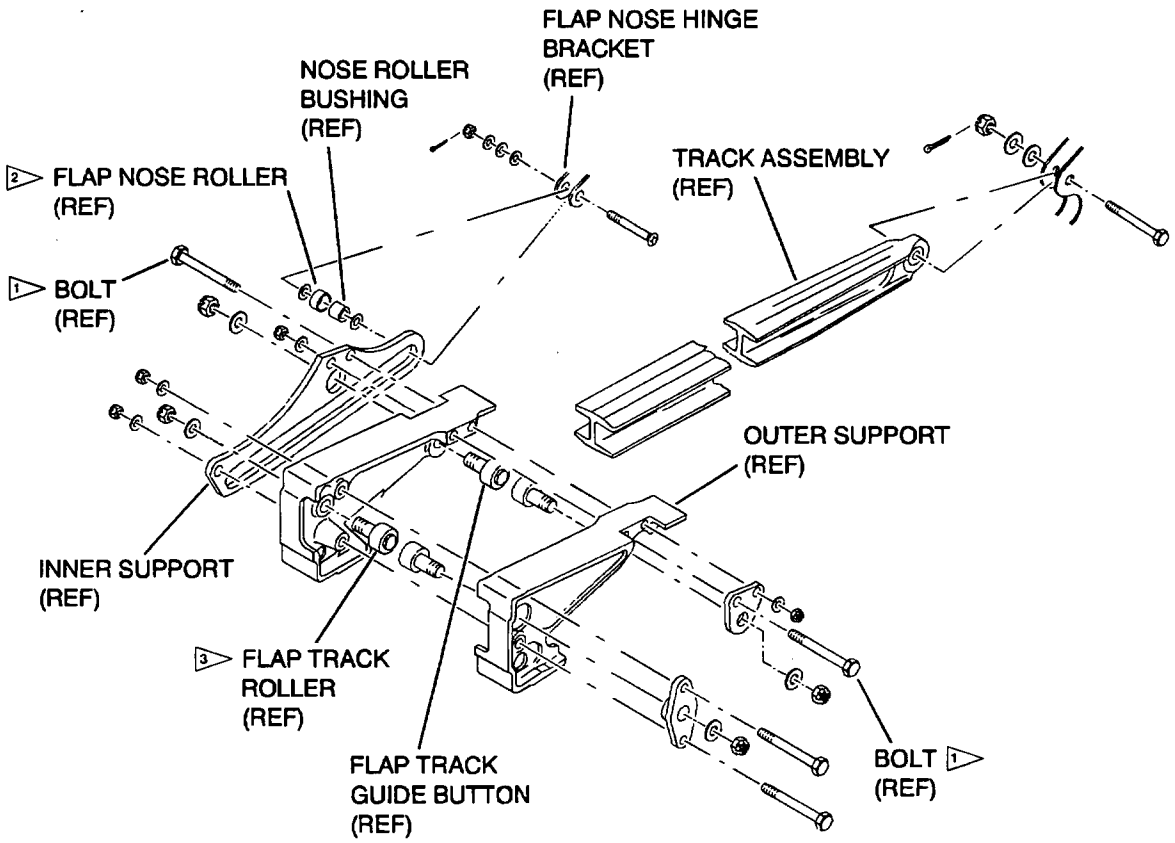


(INBOARD TRACK ASSEMBLY)

Detail C

M35-275001-201-02

Flap Installation
Figure 201 (Sheet 2 of 3)



(OUTBOARD TRACK ASSEMBLY)

Detail D

M35-275001-201-03

Flap Installation
Figure 201 (Sheet 3 of 3)

EFFECTIVITY: 35-002 THRU 35-416, 35-418, 35-420 THRU 35-476, 35-478, 35-480 THRU 35-482, AND 36-002 THRU 36-050

(5) Disconnect bonding jumper (when installed) from flap assembly. A bonding jumper is located adjacent to the outboard and the inboard track assemblies.
(6) Remove bolts securing flap to flap nose track, releasing flap nose rollers, nose roller bushings, and washers from flap nose track.
(7) Remove flap push-pull access cover and disconnect bonding jumper and push-pull tube.
(8) Remove flap from aircraft.

C. Installation of Existing Flap (See Figure 201.)
(1) Prior to installation of new flap assembly, check all components.
(a) Check all cam followers for scoring and excessive wear. Replace as necessary.
(b) Check all flap tracks for rust.
(c) Check pushrod assembly end bearings. Replace as necessary.
(d) Check bearings in flap tracks. Replace as necessary.
(e) Check flap track guide buttons, located on cam followers, for excessive wear. Replace as necessary.
(f) Check for wear marks between flap fairing and aft fuselage fairing. Repair as necessary. (Refer to 53-50-04.)
(g) *On Aircraft 35-002 thru 35-676 and 36-002 thru 36-063 not modified per SB 35/36-27-32*, inspect the hole in the flap sector support brackets for cracks and elongation. (Refer to 27-50-06.)

(2) Clean flap tracks and flap nose tracks.
(3) Lubricate cam followers with grease (Refer to Chapter 12) using a zerk gun and a 90° rubber tip adapter.
(4) Align flap and flap tracks with cam followers and push-pull tube with push-pull tube access hole in wing rear spar.
(5) Insert flap tracks on cam followers and push-pull tube into access hole, ensuring that flap tracks align on forward cam followers and flap nose track is between flap nose hinge brackets.
(6) With flap nose tracks on flap nose hinge brackets, install flap nose rollers, nose roller bushings, and washers using attaching parts.
(7) Connect flap bonding jumper to rear wing spar and push-pull tube to flap sector.
(8) Attach flap push-pull tube to sector by inserting bolt from the bottom side and secure with nut, washer, and cotter pin. Remove rigging pin from aft sector.
(9) Rig flaps. (Refer to 27-50-00, Adjustment/Test.)
(10) If required, replace protective tape on lower wing skin and upper wing skin lower surface.
(11) Install sector assembly access cover.
(12) Restore hydraulic power to aircraft.
(13) Restore electrical power to aircraft.
(14) Stow flaps and spoilers in full retract position.
(15) Restore aircraft to normal.

D. Installation of Replacement Flap (See Figure 202.)

NOTE: Attaching hardware not supplied with a new flap should be removed from the old flap and used on the new flaps.

(1) Prior to installation of new flap assembly, check all components.
(a) Check all cam followers and flap nose rollers and nose roller bushings for scoring and wear. Replace as necessary.
(b) Check all flap tracks for rust.
(c) Check pushrod assembly end bearings. Replace as necessary.
(d) Check bearings in flap tracks. Replace as necessary.
(e) Check flap track guide buttons, located on cam followers, for wear. Replace as necessary.
(f) *On Aircraft 35-002 thru 35-676 and 36-002 thru 36-063 not modified per SB 35/36-27-32*, inspect the hole in the flap sector support brackets for cracks and elongation. (Refer to 27-50-06.)

(2) Clean flap tracks and flap nose tracks.
(3) Lubricate cam followers with grease (Refer to Chapter 12) using zerk gun with a 90° rubber tip adapter.
(4) Install flap tracks on flaps using attaching parts removed from old flap.
(5) Connect bonding jumper and push-pull tube to flap.
(6) Align flap and flap tracks with cam followers and push-pull tube with push-pull tube access hole in wing rear spar.
(7) Insert flap tracks on cam followers and push-pull tube into access hole, ensuring that flap tracks align on forward cam followers and flap nose track is between flap nose hinge brackets.
(8) Raise flap by hand to the retracted position.

NOTE: Flap cannot be raised to the full retract position until the flap fairing has been trimmed.

(9) Measure and mark flap fairing so that it may be trimmed to the gap shown in Figure 202.
(10) Lower flap and trim along the trim line previously established.
(11) Raise flap by hand to the full retract position and check for proper gap between fuselage fairing and flap fairing. Trim as necessary to obtain uniform gap.
(12) After gap has been set on flap assembly, apply protective tape #8641 to lower wing skin as follows:
  (a) Clean any oxides off lower wing skin from trailing edge forward 1 inch with fine abrasive pad. Touch up chemical film. (Refer to Chapter 12.)
  (b) Clean surface with a 50/50 mixture of water and isopropyl alcohol.
  (c) After surface is dry, apply a light coat of 3M #86 adhesion promoter with a clean, lint-free cloth. Allow promoter to dry 15 to 20 minutes.
  (d) Install #8641 tape to lower wing surface aligning edge of tape with trailing edge of wing skin.
  (e) Use a roller to work out any bubbles being careful not to stretch the tape.
  (f) Clean any oxides off upper wing skin lower surface from trailing edge forward 1-1/2 inches with fine abrasive pad. Touch up chemical film. (Refer to Chapter 12.)
  (g) Clean surface with a 50/50 mixture of water and isopropyl alcohol.
  (h) After surface is dry, apply a light coat of 3M #86 adhesion promoter with a clean, lint-free cloth. Allow promoter to dry 15 to 20 minutes.
  (i) Install #8681 tape to lower wing surface aligning edge of tape with trailing edge of wing skin.
  (j) Use a roller to work out any bubbles being careful not to stretch the tape.
(13) With flap nose tracks on flap nose hinge brackets, install flap nose rollers, nose roller bushings, and washers using attaching parts.
(14) Connect flap bonding jumper to rear wing spar.
(15) Attach flap push-pull tube to sector by inserting bolt from the bottom side and secure with nut, washer, and cotter pin.
(16) Rig flaps. (Refer to 27-50-00, Adjustment/Test.)
(17) Perform functional test of the stall warning system (27-31-00, Adjustment/Test), operational checks of the spoiler warning system (27-60-00, Inspection/Check), and functional test of the landing gear aural warning system (Chapter 32), and make adjustments as required.

2. Approved Repairs
   A. Tools and Equipment

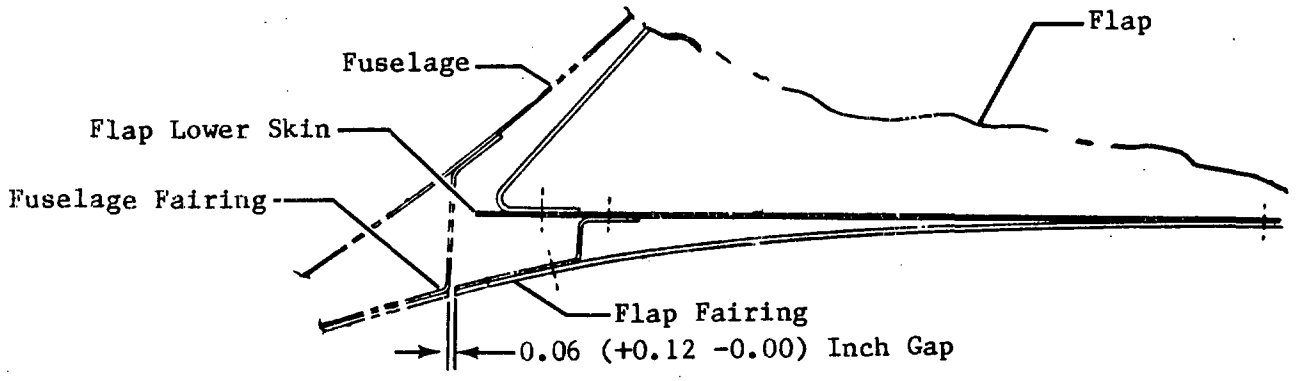
NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Adhesive	680	Loctite Co. Newington, CT	Bearing retention.

B. Replacement of Flap Track Assembly Bearings

NOTE: The following procedure is to be used if track assembly bearings are damaged, worn, or become loose in the track assembly.

(1) Remove attaching parts and track assembly from flap.
(2) Remove and discard old bearing.
(3) Clean bearing hole of all foreign matter, degrease, and wipe dry.
(4) Apply adhesive and install bearing in track assembly. Use in accordance with manufacturer's instructions.
(5) After adhesive has cured, install track assembly on flap and secure with attaching parts.



Detail A

Replacement Flap Installation
Figure 202

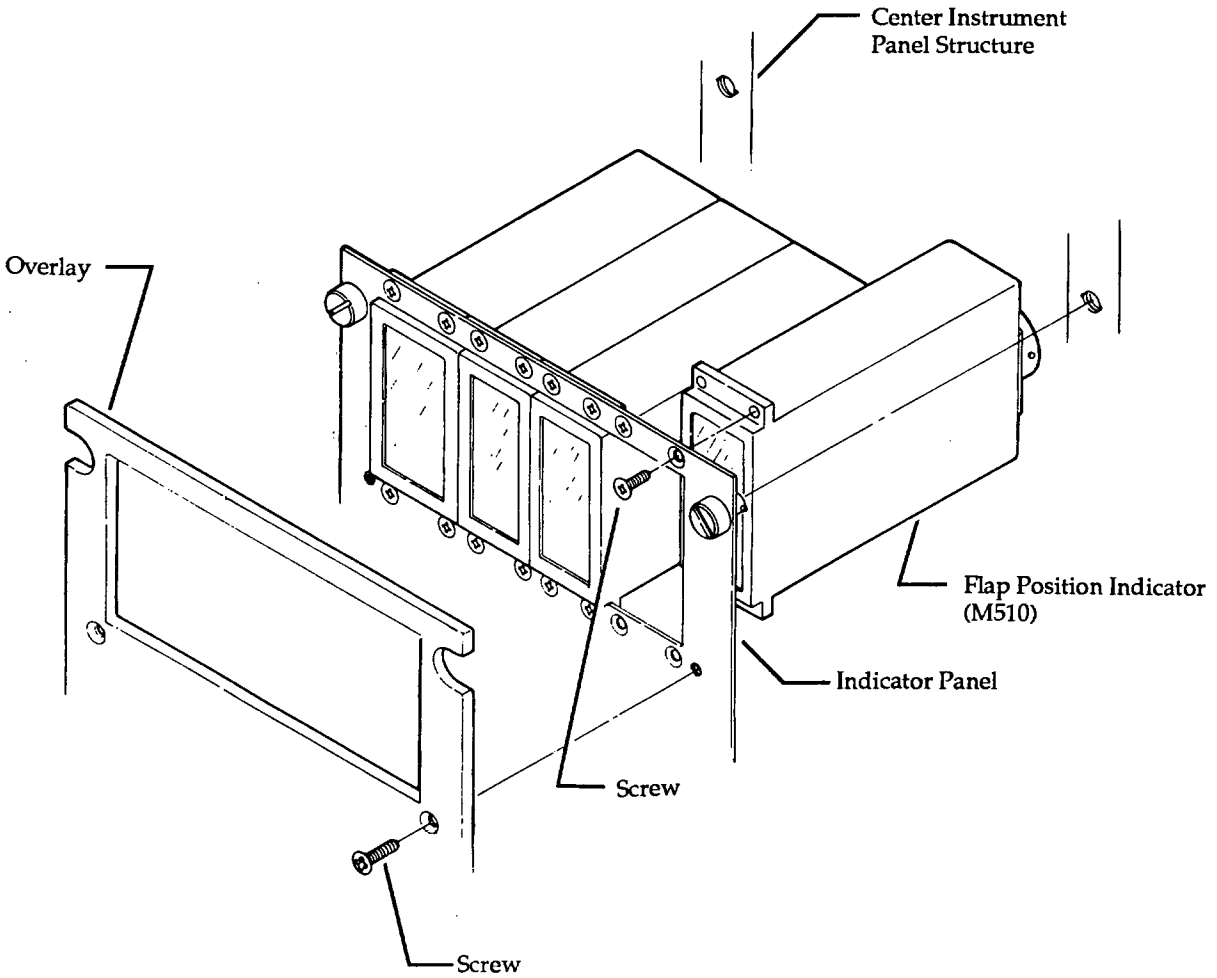
FLAP POSITION INDICATOR - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Remove Flap Position Indicator (See figure 201.)
      (1) Remove electrical power from aircraft.
      (2) Remove attaching parts, overlay, and indicator panel from instrument panel.
      (3) Disconnect electrical connector from flap position indicator.
      (4) Remove attaching parts and flap position indicator from indicator panel.
   B. Install Flap Position Indicator (See figure 201.)
      (1) Position flap position indicator in indicator panel and secure with attaching parts.
      (2) Connect electrical connector to flap position indicator.
      (3) Position overlay and indicator panel in instrument panel and secure with attaching parts.
      (4) Restore electrical power to aircraft.
      (5) Perform flap system functional test. (Refer to 27-50-00, Adjustment/Test.)

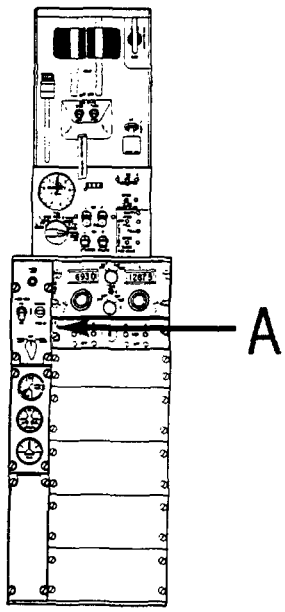
2. **Adjustment/Test** (See figure 202.)
   A. Calibrate flap position indicator as follows:

      NOTE:
      - The flap position indicator, located on the center instrument panel, indicates flap position. Input to the indicator is from a position transmitter attached to the left-hand flap sector.
      - Calibration of the flap position indicator is performed at the position indicator calibration assembly located inside the trim switch panel.

      (1) Rig flaps in accordance with procedures outlined in Flap System Adjustment/Test. (Refer to 27-50-00, Adjustment/Test.)
      (2) Remove equipment as necessary from the pedestal adjacent to the trim switch panel. (Refer to Chapter 31.) This allows access to the position indicator calibration assembly.
      (3) Actuate flaps to full up position.
      (4) Adjust zero flap potentiometer to obtain UP indication on the flap position indicator.
      (5) Actuate flaps to full down position.
      (6) Adjust flap down potentiometer to obtain DN indication on flap position indicator.
      (7) Cycle flaps while checking respective indication on flap position indicator.
      (8) Readjust flap down potentiometer as necessary to obtain coordinated indications.
      (9) Replace and secure equipment in the pedestal. (Refer to Chapter 31.)
      (10) Perform flap system functional test. (Refer to 27-50-00, Adjustment/Test.)



Flap Position Indicator Installation
Figure 201



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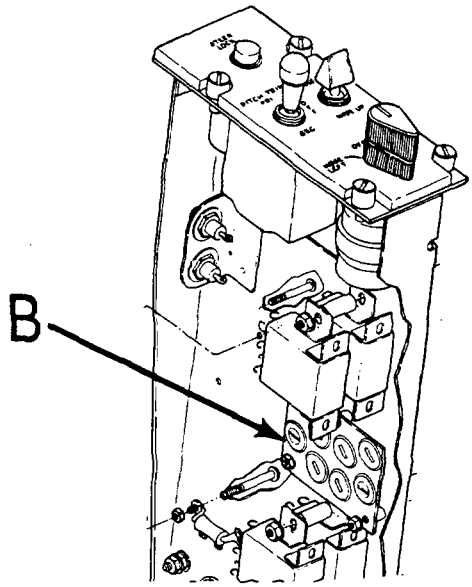
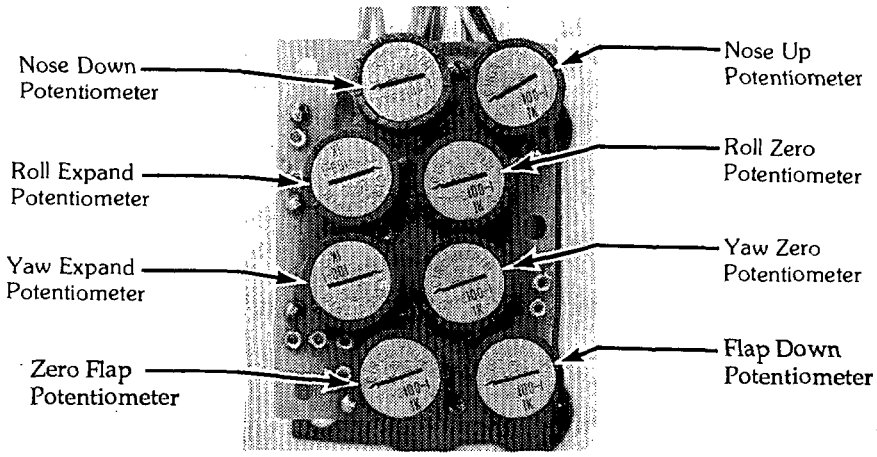


Detail A



Detail B

Position Indicator Calibration Assembly
Figure 202

FLAP CONTROL VALVE - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

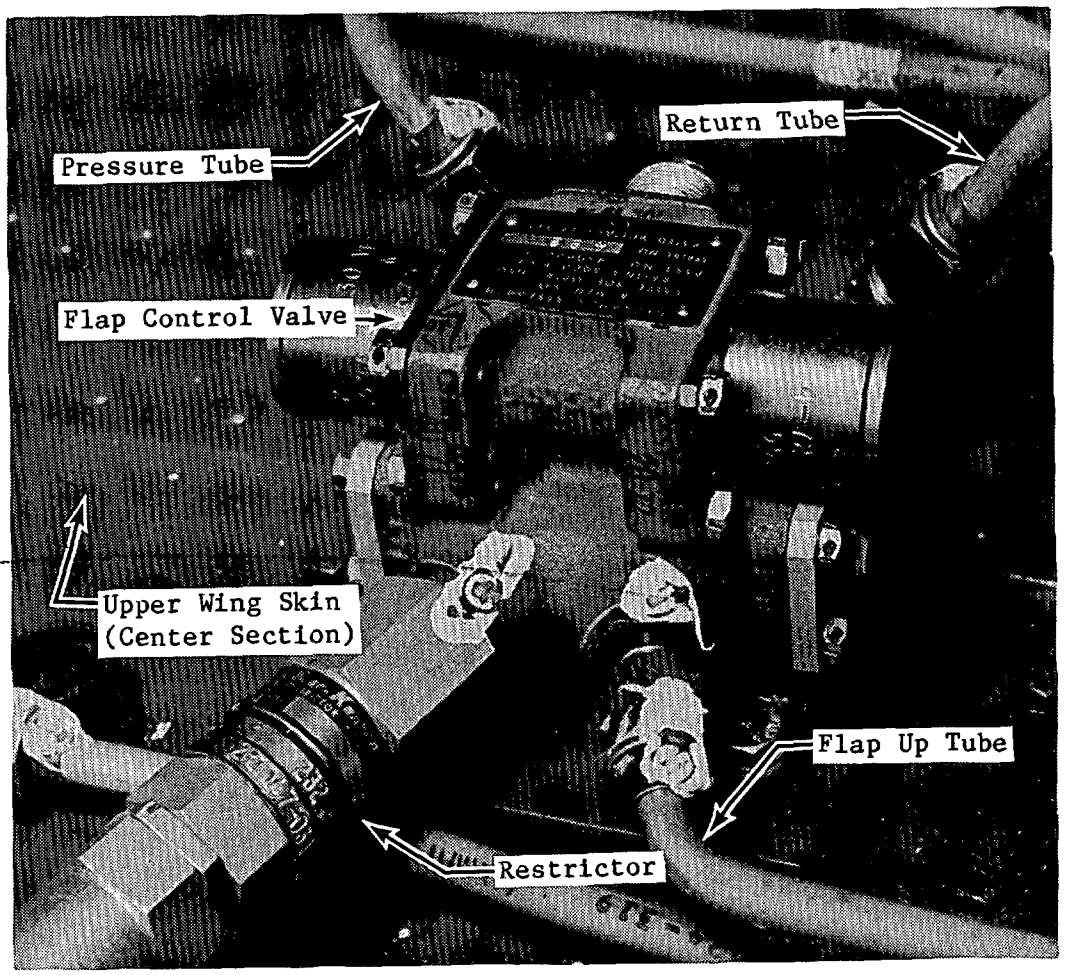
NOTE: The flap control valve is a 4-way, 3-position solenoid-operated valve which directs pressure to each of the flap actuators to extend or retract the flaps. This valve operates on 28 vdc through the FLAPS circuit breaker. The valve is located on the top skin of the wing between wing spars No. 7 and No. 8 approximately eleven inches to the left of the centerline of the wing. Access is through the main landing gear wheel well.

A. Remove Flap Control Valve (See figure 201.)

(1) Depressurize hydraulic system.
(2) Pull FLAP circuit breaker.
(3) Remove access covers and disconnect electrical connector from valve.
(4) Disconnect pressure tube from elbow at pressure port of valve.
(5) Disconnect flap valve return tube and brake return line from tee at return port of valve.
(6) Disconnect flap-up tube assembly from union at flap-up port of valve.
(7) Disconnect restrictor from flap-down port of valve.
(8) Remove four bolts attaching valve to mounting bracket and remove valve from aircraft.
(9) Remove elbow, tee, restrictor, and union from ports of valve.

B. Install Flap Control Valve (See figure 201.)

(1) Inspect elbow, tee, restrictor, and union removed from valve and, if serviceable, install new valve using new O-ring and packings.
(2) Position valve on mounting bracket and check that elbow and tee are clocked to allow connection of hydraulic tubes.
(3) Attach valve to mounting bracket using bolts and lock washers.
(4) Connect hydraulic tubes to ports from which they were removed.
(5) Connect electrical connector to valve and safety wire. Install access covers.
(6) Depress FLAP circuit breaker.
(7) Pressurize hydraulic system and check flap operation.



ORIGINAL
As Received By
ATP

Flap control Valve Installation
Figure 201

FLAP ACTUATOR - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

NOTE: ° Two flap actuators are installed on the aircraft. The following procedure is applicable to either actuator.

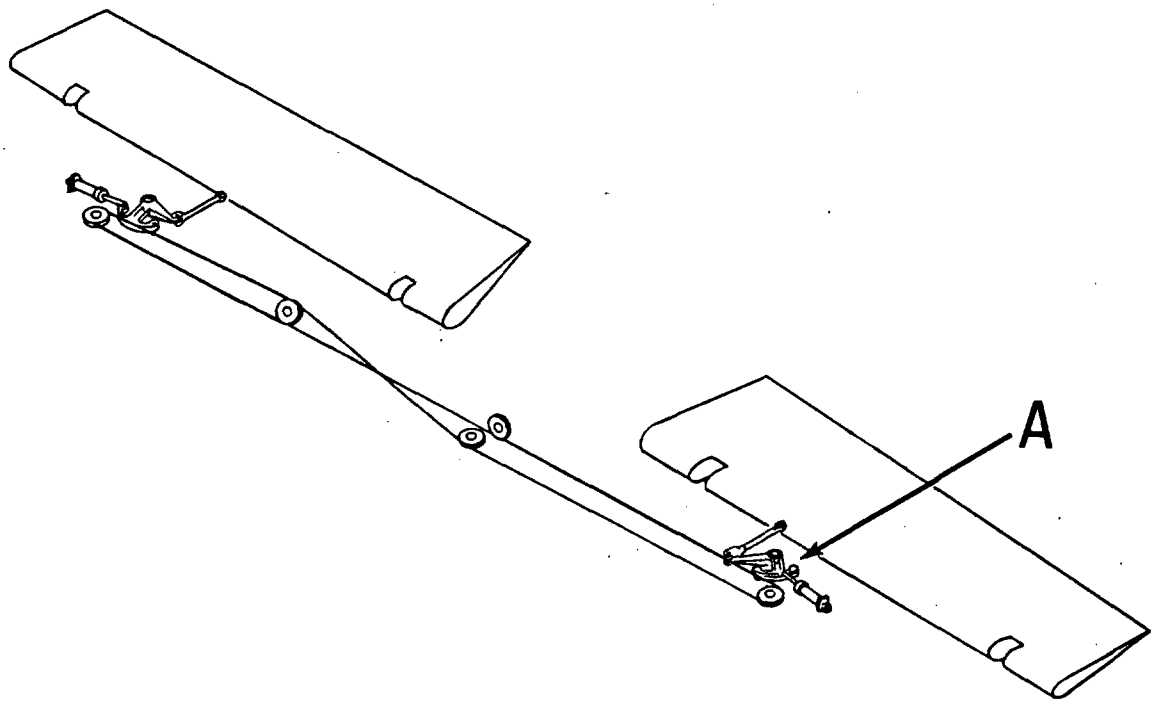
° Access to the flap actuator is through the flap actuator access cover on the lower wing skin.

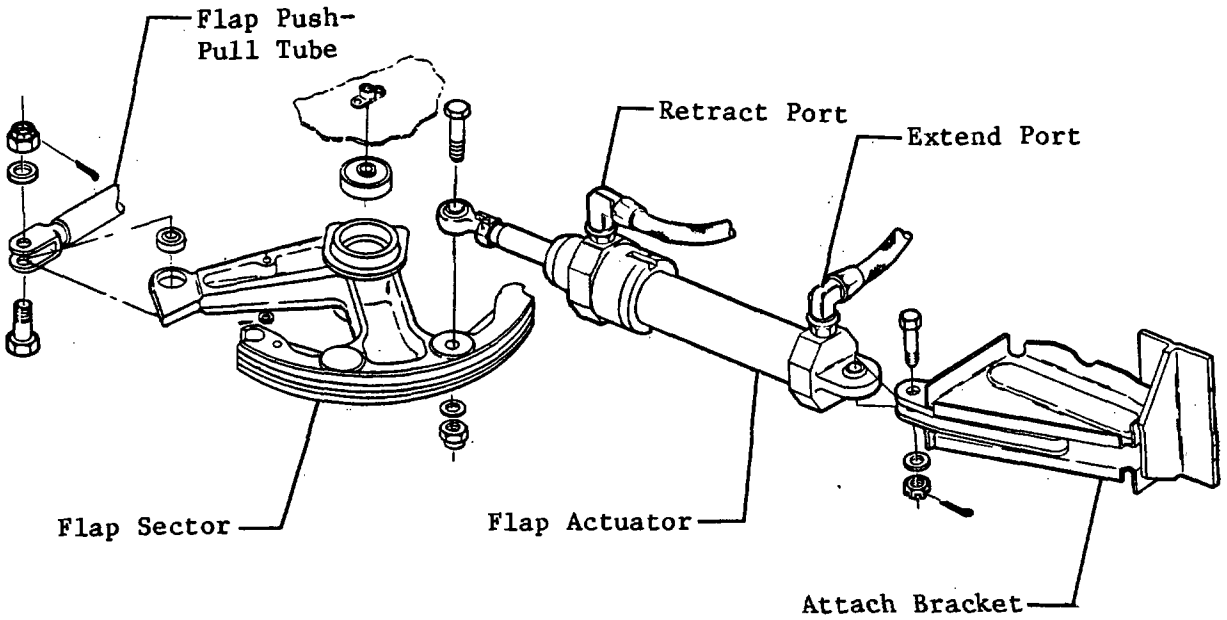
A. Remove Flap Actuator (See figure 201.)

(1) Depressurize the hydraulic system. (Refer to Chapter 29.)
(2) Remove flap actuator access cover.
(3) Disconnect and cap hydraulic lines at actuator.
(4) Disconnect actuator from attach bracket and sector and remove actuator from aircraft.

B. Install Flap Actuator (See figure 201.)

(1) Position flap actuator between attach bracket and sector and secure with attaching parts.
(2) Remove caps and connect hydraulic lines to actuator.
(3) Rig flap system. (Refer to 27-50-00, Adjustment/Test.)
(4) Bleed hydraulic system. (Refer to Chapter 29.)
(5) Install access cover.





Detail A

Flap Actuator Installation
Figure 201

FLAP ACTUATOR - APPROVED REPAIRS

1. DISASSEMBLY

NOTE: ° The following procedure lists disassembly and assembly of flap actuator necessary to replace O-rings, backup rings, and felt wiper. New O-rings, backup rings, and felt wiper must be installed at each assembly. These new items are to be immersed in MIL-H-5606 hydraulic fluid immediately prior to installation.

° All O-ring grooves are to be free of gouges, scratches, and burrs. Assure that all items are thoroughly clean prior to installation.

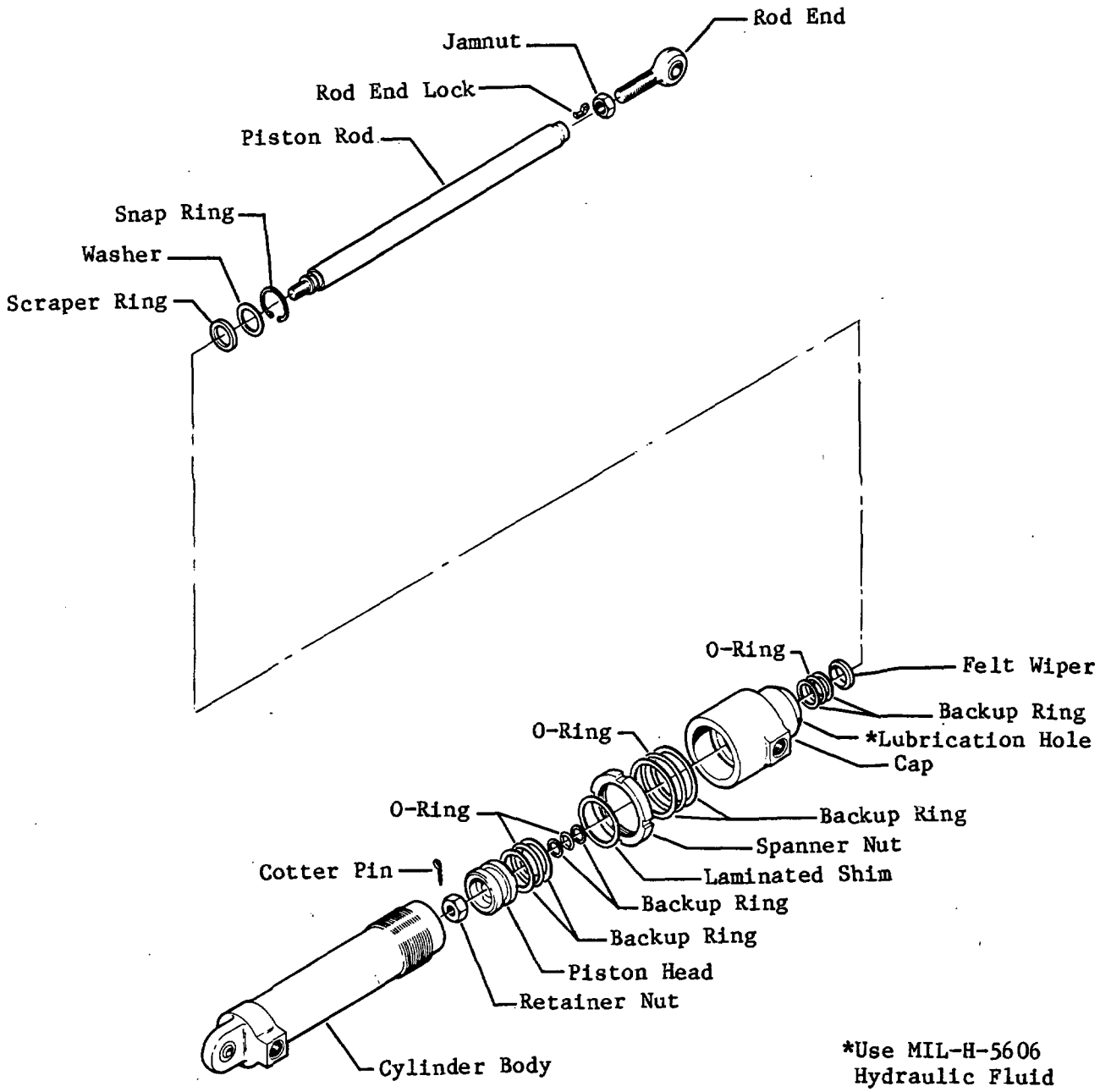
A. Disassemble Flap Actuator (See figure 801.)

(1) Cut safety wire and loosen jamnut sufficiently to release rod end lock. Remove rod end from piston rod.
(2) Cut safety wire and loosen spanner nut. Remove cap from cylinder body. Remove piston rod from cylinder body.
(3) Remove cotter pin and retainer nut securing piston head to piston rod.
(4) Remove snap ring from cap. Remove all backup rings, O-rings, washer, laminated shim, scraper ring, and felt wiper from piston rod, piston head, cylinder body, and cap.
(5) Clean all parts in Stoddard solvent or equivalent. Dry parts with clean, dry, compressed air.
(6) Inspect cylinder body, cap, piston rod, and piston head for nicks, scratches, or excessive wear. Buff out minor defects with an oil stone, then polish with crocus cloth.

2. ASSEMBLY

A. Assemble Flap Actuator (See figure 801.)

(1) Immerse all new items in MIL-H-5606 hydraulic fluid.
(2) Assemble O-ring and backup rings on piston rod.
(3) Install piston head on piston rod and secure with retainer nut and cotter pin. Assemble O-ring and backup rings on piston head.
(4) Apply a thin coat of MIL-H-5606 hydraulic fluid to piston head and insert piston rod with piston head attached into cylinder body.
(5) Assemble O-rings, backup rings, laminated shim, and felt wiper in cap. Screw cap on cylinder body as far as possible, assuring that the extend and retract ports are in line.
(6) Lubrication of the felt wiper is required every 600 hours. Use only MIL-H-5606 hydraulic fluid.
(7) Tighten spanner nut against cap and safety wire.
(8) Assemble scraper ring, washer, and snap ring in cap. Install rod end with jamnut and rod end lock in piston rod.
(9) Connect actuator to a hydraulic test bench. Pressure check actuator, both extend and retract positions, to 2250 psi for 3 minutes. Leakage should not exceed one drop in the 3-minute period.
(10) With the actuator retracted, adjust for 12.950 (±0.010) inches between centerline of actuator bearing and rod end. Safety wire rod end lock.



Flap Actuator Assembly
Figure 801

FLAP POSITION SWITCH - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

NOTE: ° On Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not equipped with AMK 76-4 (Reduced Approach Speed System), there are four cam roller type limit switches, two 25° position switches and two 13° position switches. As flap sector turns a cam in the middle of the sector, it actuates the 25° position switches and the perimeter of the sectors actuates the 13° position switches.

° On Aircraft 35-067 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482, 36-018 thru 36-050 and prior Aircraft equipped with AMK 76-4 (Reduced Approach Speed System), there are two, three-layer, rotary-type position switches. The position switches provide flap position information at 3°, 13°, and 25° for the stall warning, spoiler, gear aural warning, and autopilot systems. The left position switch is connected to the sector by a bellcrank and linkage. As the sector turns, it actuates the position switch. The left position switch assembly includes the position switch attached to its mounting bracket with a clamp and the bellcrank along with the attaching parts necessary to complete the assembly. The right position assembly includes the position switch attached to its mounting bracket with a clamp and the actuating lever along with the attaching parts necessary to complete the assembly. The position switches are preset at the factory. When switch assemblies are replaced, some minor adjustments may be required. If adjustments are required, refer to Adjustment/Test this section.

A. Remove Flap Position Switch (Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not equipped with AMK 76-4 [Reduced Approach Speed System])

NOTE: Removal and installation of the flap position switches are identical for both RH and LH installations.

WARNING: BLOCK THE SPOILERS UP TO PREVENT INADVERTENT SPOILERS RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps to gain access to the position switch. Block spoilers.
(2) To aid in proper installation, note the position of the switch to be removed. (See figure 201.)
(3) Cut and tag electrical wiring.
(4) Remove attaching hardware and remove position switch from the mounting bracket.

B. **Install Flap Position Switch** (Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not equipped with AMK 76-4 [Reduced Approach Speed System])

NOTE: Removal and installation of the flap position switches are identical for both RH and LH installations.

WARNING: BLOCK THE SPOILERS UP TO PREVENT INADVERTENT SPOILERS RETRACTION AND BODILY INJURY.

(1) Place the position switch in position, as noted in step A.(2).
(2) Secure the position switch in place with attaching hardware.
(3) Remove tags and connect electrical wires to the switch.
(4) Remove blocks from the spoilers and return the flaps and spoilers to the neutral position.
(5) Perform Flap Position Switch Adjustment/Test procedures.

C. **Remove Flap Position Switch Assembly** (Aircraft 35-067 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482, 36-018 thru 36-050, and prior Aircraft equipped with AMK 76-4 [Reduced Approach Speed System])

WARNING: BLOCK THE SPOILERS UP TO PREVENT INADVERTENT SPOILERS RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps to gain access to flap position switches. Block spoilers.
(2) Remove cotter pin, washer, and pin from the point where the lever (RH), or bellcrank (LH), attaches to the link from the flap sector.
(3) On the LH position switch, also remove the cotter pin, washer, and pin from the point where the bellcrank is attached to the flap position potentiometer link.
(4) Remove screws, nuts, and washers attaching the switch assembly to wing spar 8.
(5) Cut and tag the switch wires.

D. **Install Flap Position Switch Assembly** (Aircraft 35-067 thru 35-416, 35-418, 35-420 thru 35-476, 35-478, 35-480 thru 35-482, 36-018 thru 36-050, and prior Aircraft equipped with AMK 76-4 [Reduced Approach Speed System])

WARNING: BLOCK THE SPOILERS UP TO PREVENT INADVERTENT SPOILERS RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps to gain access to flap limit switch location. Block spoilers.
(2) Attach flap position switch assembly to wing spar 8.
(3) Install the pin, washer, and cotter pin at the point where the lever (RH), or bellcrank (LH), attaches to the link of the flap sector.
(4) On the LH position switch, install pin, washer, and cotter pin where the position switch assembly bellcrank attaches to the flap position potentiometer link.
(5) Remove tags and connect wires to the switch assembly.
(6) Perform the Flap Position Switch Adjustment/Test procedures.

2. **ADJUSTMENT/TEST**

A. Adjustment of Flap Position Switch (Aircraft 35-002 thru 35-066 and 36-002 thru 36-017, not equipped with AMK 76-4 [Reduced Approach Speed System])

NOTE: The flaps must be properly rigged prior to adjusting the flap position switches.

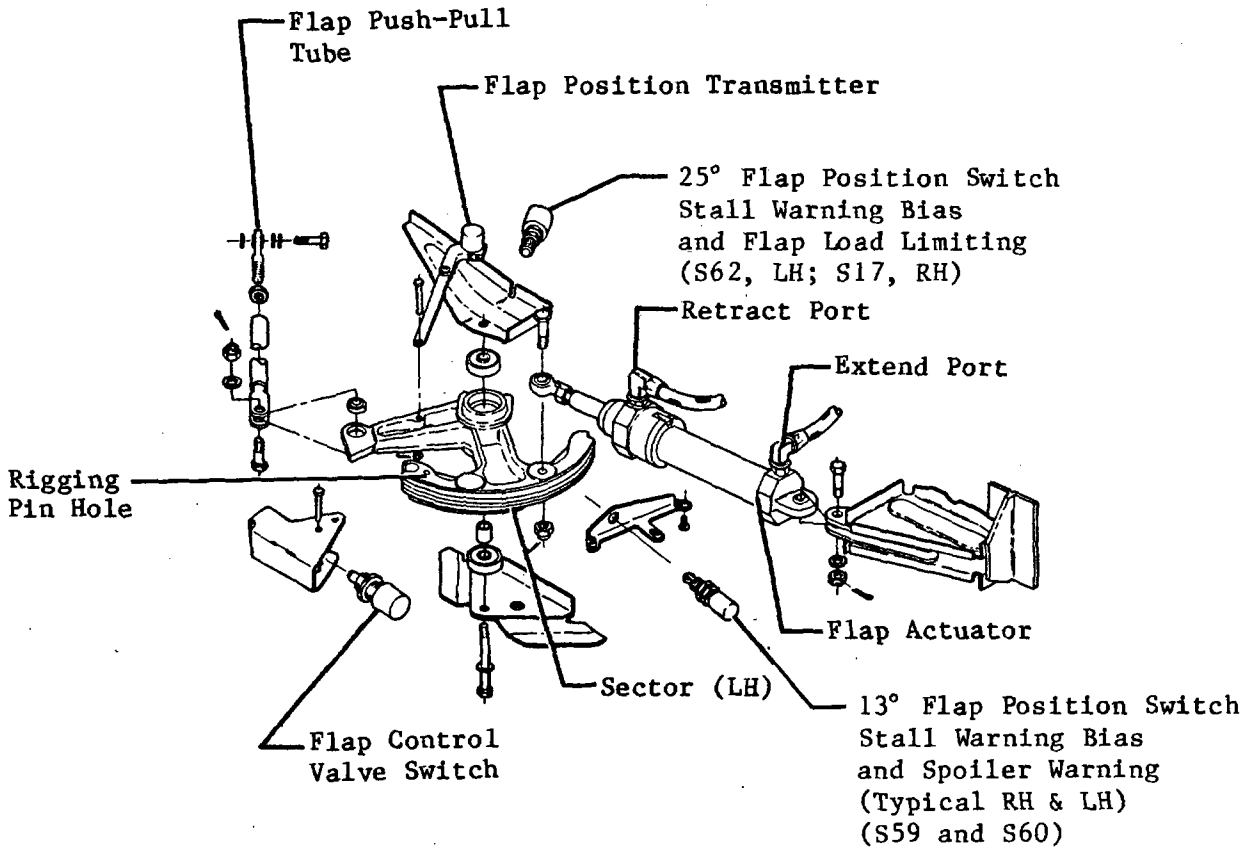
(1) Gain access to the stall warning bias box by removing the nose compartment access doors.
(2) Disconnect plug P536 from the stall warning bias box.
(3) Attach clinometer to flap.
(4) Using an ohmmeter, probe between pin E of plug P536 and ground; observe open circuit.
(5) Move flaps to 13° down; observe that ohmmeter indicates continuity.
(6) If continuity is not indicated in step (5), adjust RH 13° flap position switch until continuity is indicated and repeat steps (4) and (5).
(7) Using an ohmmeter, probe between pin D of plug P536 and ground with flaps in neutral position. Observe open circuit.
(8) Move flaps to 13° down; observe that ohmmeter indicates continuity.
(9) If continuity is not indicated in step (8), adjust LH 13° flap position switch, then move the flaps to 25° and observe an open circuit indicated on ohmmeter.
(10) Probe pin J to ground; observe continuity with flaps at 13°, then move flaps to 25° and observe an open circuit indicated on ohmmeter.
(11) If an open circuit is not indicated at 25° flaps in step (10), adjust RH 25° position switch until an open circuit exists and repeat step (10).
(12) With flaps at 13°, probe between pin H of plug P536 and ground. Observe that ohmmeter indicates continuity, then move flaps to 25° and observe that an open circuit is indicated on ohmmeter.
(13) If an open circuit is not indicated at 25° flaps in step (12), adjust LH 25° position switch until an open circuit exists and repeat step (12).
(14) Connect plug P536 to stall warning bias box and install nose compartment doors.

B. Adjustment of Flap Position Switch (Aircraft 35-067 thru 35-416, 35-418, 35-420 thru 35-476, 35-480 thru 35-482, 36-018 thru 36-050, and prior Aircraft equipped with AMK 76-4 [Reduced Approach Speed System]) (See figure 202.)

NOTE: The flaps must be properly rigged prior to adjusting flap position switches.

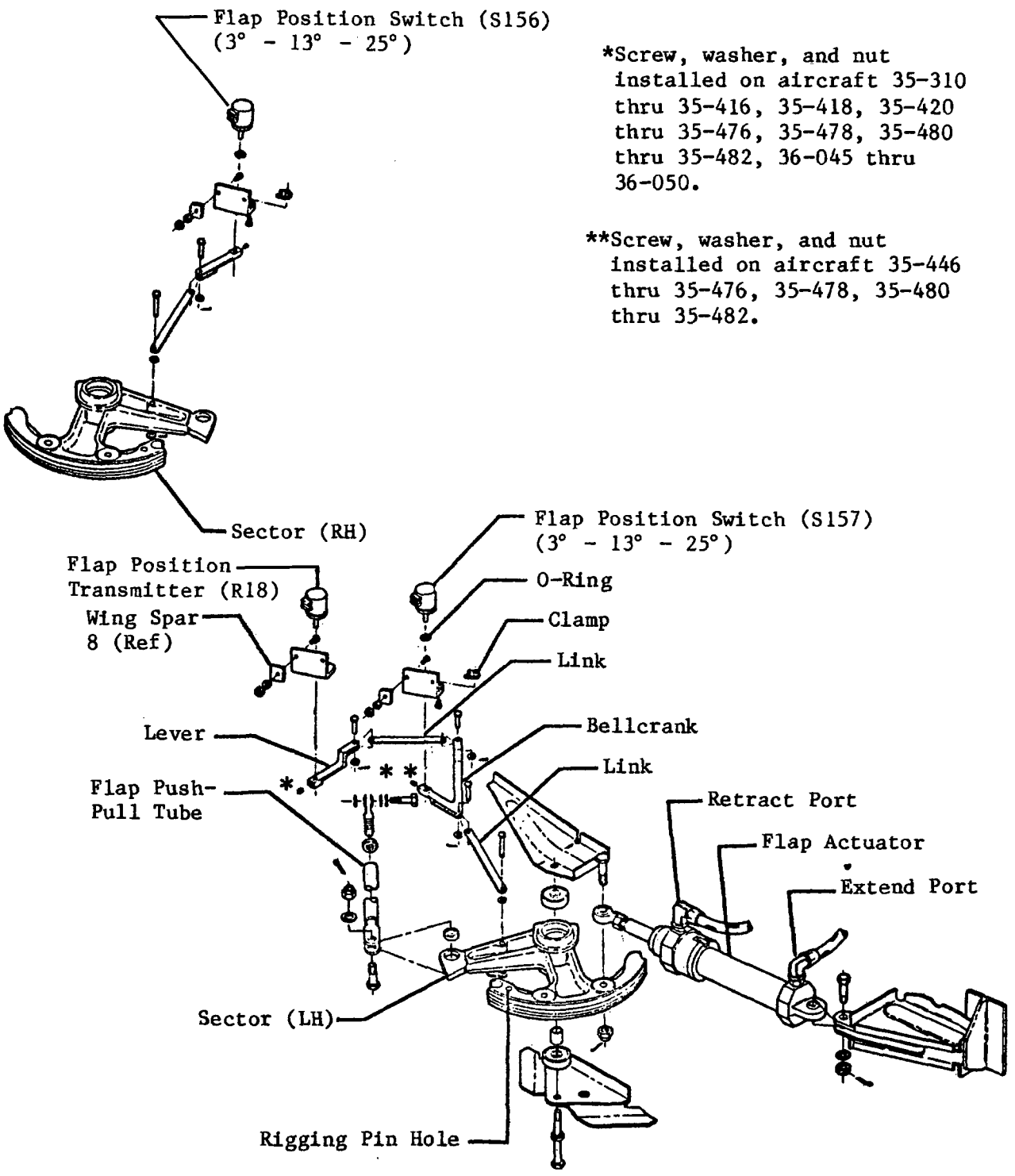
(1) Gain access to stall warning computer by removing nose compartment access doors.
(2) Disconnect plugs P914 and P913 from stall warning computer.

(3) Attach a clinometer to flaps.
(4) Probe between pin U of plug P913 and ground; observe an open circuit indication on ohmmeter. Then move flaps to 3° (±0.5°) and observe a continuity indication on ohmmeter.
(5) Repeat step (4) for pin U of plug P914 and ground for RH flap position switch.
(6) If indications in steps (4) or (5) are correct, proceed to step (7). If indications are not correct, proceed as follows:
    (a) Connect VOM between terminals NO and C of applicable switch assembly (Deck A).
    (b) Loosen lock screw on adjustment screw A and rotate adjustment screw CW or CCW until VOM indicates continuity. Tighten lock screw and remove VOM.
    (c) Repeat steps (4) or (5) to verify adjustment.
(7) Probe between pin E of plug P913 and ground; observe an open circuit indication on ohmmeter. Then move flaps to 13° (±0.5°) and observe a continuity indication on ohmmeter.
(8) Repeat step (7) for pin E of plug P914 and ground for RH flap position switch.
(9) If indications in steps (7) or (8) are correct, proceed to step (10). If indications are not correct, proceed as follows:
    (a) Connect VOM between terminals NO and C of applicable switch assembly (Deck B).
    (b) Loosen lock screw on adjustment screw A and rotate adjustment screw CW or CCW until VOM indicates continuity. Tighten lock screw and remove VOM.
    (c) Repeat steps (7) or (8) to verify adjustment.
(10) Probe between pin F of plug P913 and ground; observe an open circuit indication on ohmmeter. Then move flaps to 25° (±0.5°) and observe a continuity indication on ohmmeter.
(11) Repeat step (11) for pin F of plug P914 and ground for RH flap position switch.
(12) If indications in steps (10) or (11) are correct, proceed to step (13). If indications are not correct, proceed as follows:
    (a) Connect VOM between terminals NO and C of applicable switch assembly (Deck C).
    (b) Loosen lock screw on adjustment screw A and rotate adjustment screw CW or CCW until VOM indicates continuity. Tighten lock screw and remove VOM.
    (c) Repeat steps (10) or (11) to verify adjustment.
(13) Connect plugs P913 and P914 to stall warning computer and install nose compartment access doors.

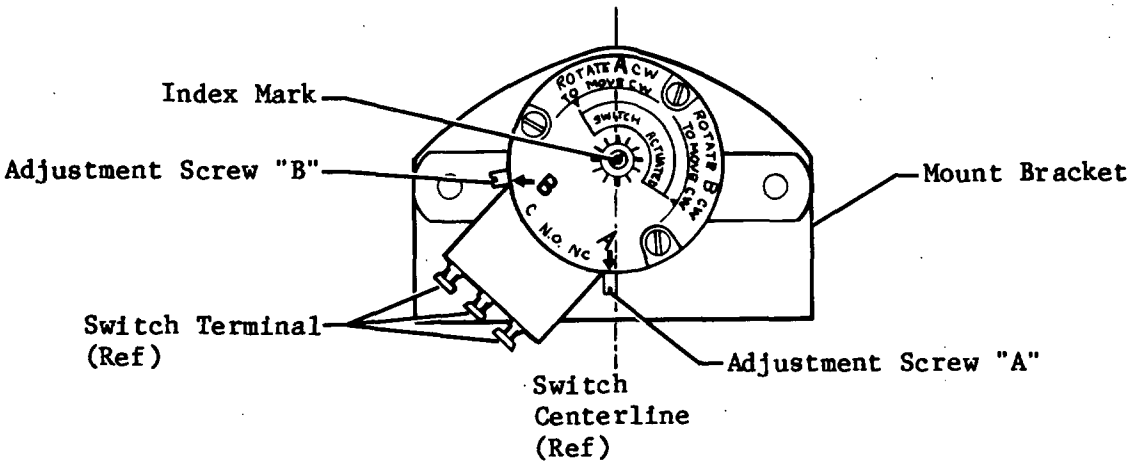


Flap Position Switch Installation
Figure 201 (Sheet 1 of 2)

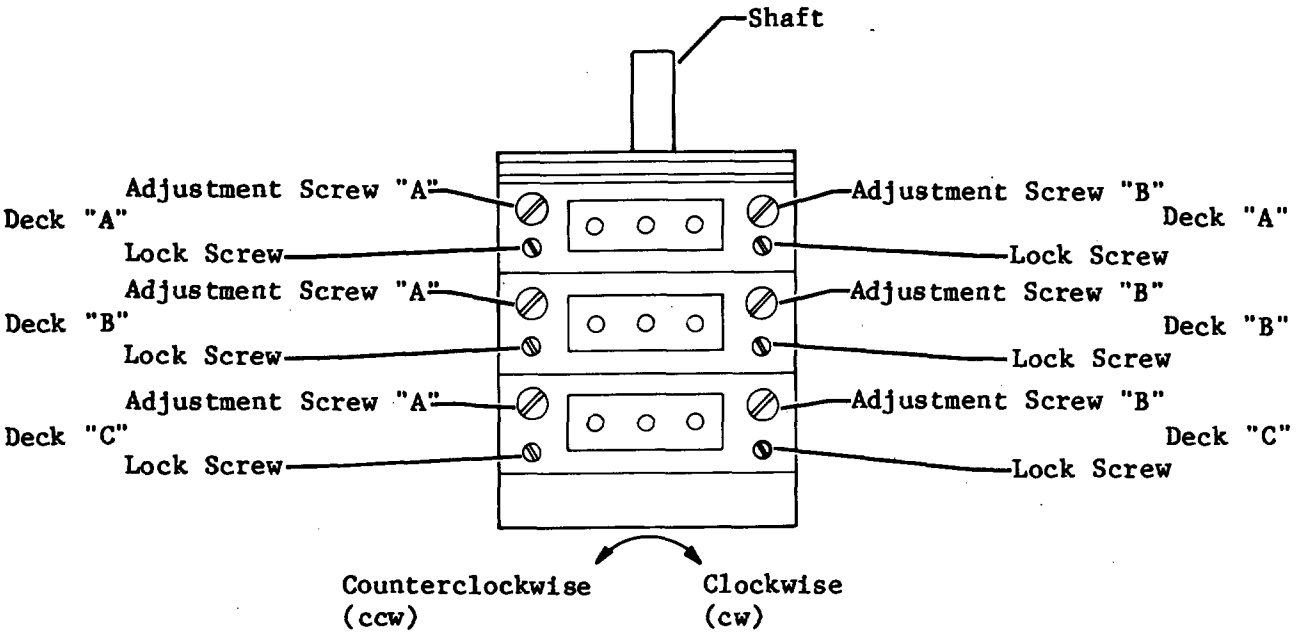
EFFECTIVITY: **35-002 thru 35-066, and 36-002 thru 36-017, Not Modified per AMK76-4, "Reduced Approach Speed System"**
MM-99
D538



Flap Position Switch Installation
Figure 201 (Sheet 2 of 2)



End View of Switch



Top View of Switch Looking Down

Flap Position Switch Adjustment
Figure 202

FLAPS SECTOR - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Removal of Flap Sector (See Figure 201.)
      (1) Connect Electrical power source to aircraft.
      (2) Connect external hydraulic power source to aircraft.
      (3) Extend flaps and spoilers.
      (4) Remove flap sector access cover from LH and RH wings.
      (5) Disconnect flap actuator from flap sector. (Refer to 27-50-04, Removal /Installation.)
      (6) Disconnect flap push-pull tube from flap sector. (Refer to 27-50-01, Removal /Installation.)
      (7) Remove safety wire from flap sector bolt, remove bolt and flap sector from wing.

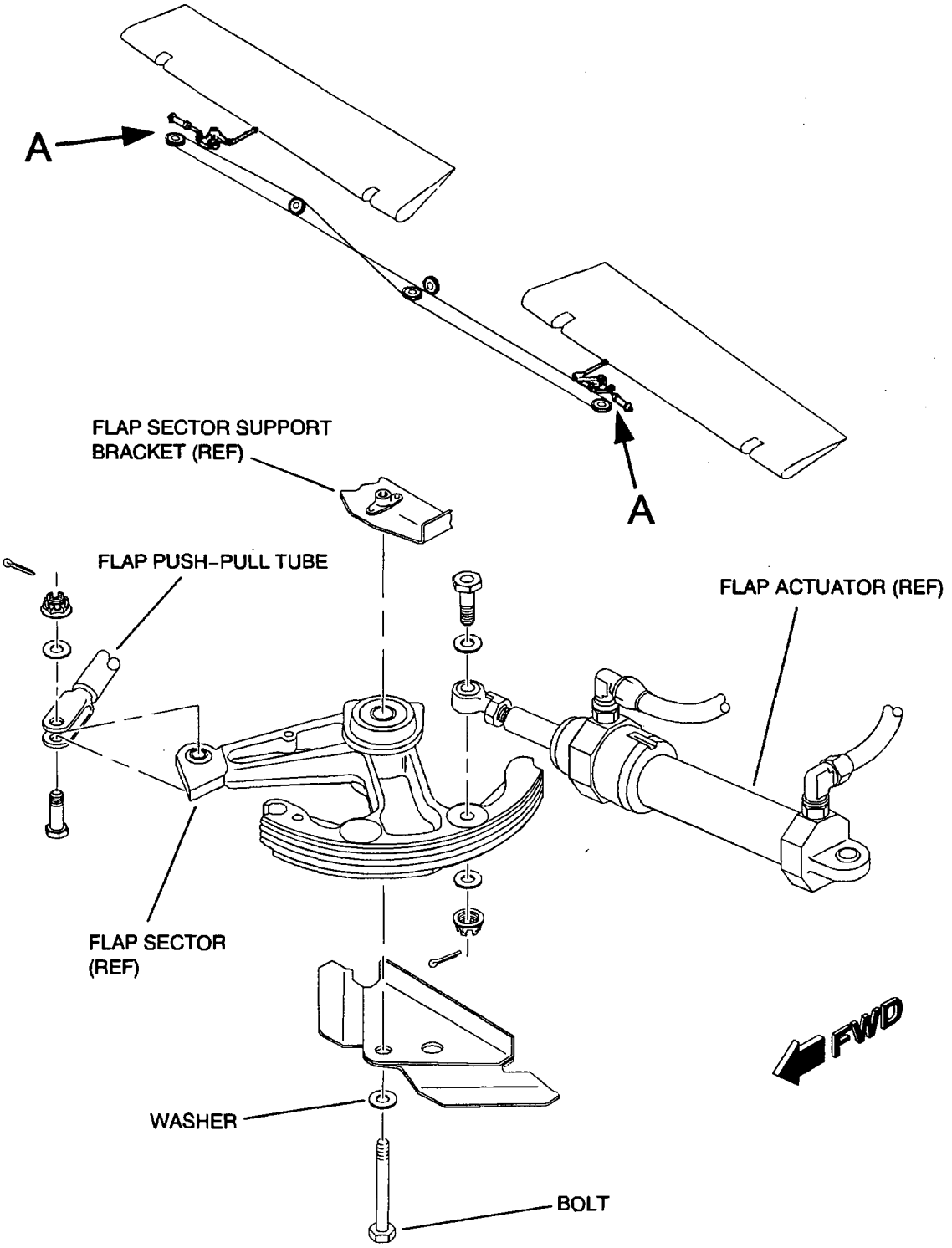
      NOTE: It is not necessary to loosen flap cable turnbuckles to remove flap sectors. Turn flap sectors over and remove cables from flap sectors.

   B. Installation of Flap Sector (See Figure 201.)
      (1) Tools and Equipment

      NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Safety Wire	MS20995C32	Commercially Available	To safety sector bolt.

      (2) Prior to installation of flap sector, check all components for excessive wear.
      (3) Install flap sector with existing hardware and safety bolt.
      (4) Connect flap actuator to flap sector. (Refer to 27-50-06, Removal/Installation.)
      (5) Connect push-pull tube to flap sector. (Refer to 27-50-01, Removal/Installation.)
      (6) Perform Flap System Functional Test. (Refer to 27-50-00, Adjustment/Test.)
      (7) Install flap sector access covers on LH and RH wings.
      (8) Disconnect external hydraulic power source from aircraft.
      (9) Disconnect electrical power source from aircraft.



(LH SHOWN; RH OPPOSITE)

Detail A

M31-275009-201-01

Flap Sector Installation
Figure 201

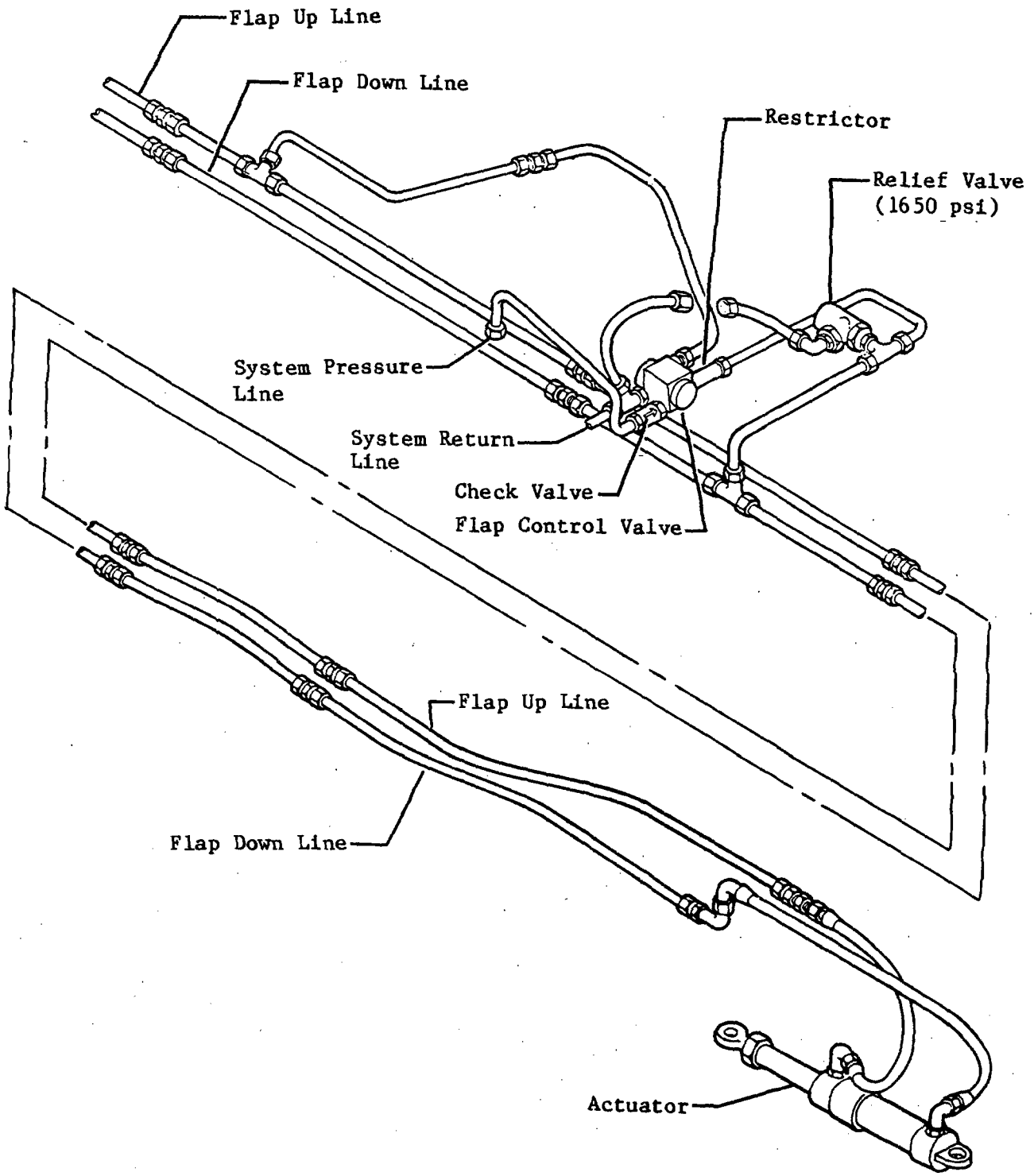
FLAP SYSTEM - DESCRIPTION AND OPERATION

1. **Description** (See Figures 1, 2, 3, and 4.)
   A. The flap system is hydraulically-actuated and electrically-controlled, providing flap settings of UP (0°), 8°, 20°, and DN (40°).
   B. The flap system consists of a left-hand and right-hand flap assembly, a flap preselect assembly and preselect switch, two position switches, two limit switches, a flap control valve switch, flap control valve, flap actuator, flap position potentiometer, flap position potentiometer, flap position indicator, and flap cables and pulleys installations. On *Aircraft 35-675 and Subsequent; 36-064 and Subsequent,* flap position indicator relay (K7104) is installed.
   C. Interconnecting cables and pulleys synchronize flap movement (within 2° of each other) throughout range of flap travel.
   D. Component Description
      (1) The single-slotted flaps are attached to the wing rear spar via tracks, rollers, and hinges.
      (2) The flap preselect assembly, located in the forward pedestal assembly, allows flaps to be adjusted to any of four settings: UP (0°), 8°, 20°, and DN (40°).
      (3) The flap preselect switch is a rotary-type switch, mounted in the forward pedestal assembly, linked to the Flap Switch. The preselect switch controls a solenoid-operated hydraulic control valve that meters hydraulic pressure to the flap actuators.
      (4) A flap position switch (S157, LH and S156, RH) is mechanically connected to each flap sector assembly. These switches provide flap position information to: aural warning, stall warning, aileron augmentation, spoiler warning, autopilot, and landing gear systems.
      (5) A flap limit switch (S84, LH and S86, RH) is mechanically connected to each flap sector assembly. These switches automatically maintain the flap position selected by the flap preselect assembly.
      (6) On *Aircraft 35-002 thru 35-531 and 36-002 thru 36-053 not modified per SB35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference,"* a flap control valve switch (S16) is installed forward of the LH flap sector assembly, and is utilized in de-energizing the flap control valve to stop flap movement when flaps reach full retract position.
      (7) The flap control valve is a 4-way, 3-position, solenoid-operated valve which directs hydraulic pressure to each flap actuator, extending or retracting the flaps. The valve operates on 28 vdc through the FLAPS circuit breaker and is controlled by the interaction of the flap preselect switch, flap limit switches, and the flap up-limit (0°) switch. The flap control valve is located on the wing top skin, between spars 7 and 8, at approximately LWS 11. Access to the flap control valve is obtained through the main landing gear wheel well.
      (8) The flap actuators are attached to the wing structure and to the flap sectors, between wing spars 7 and 8 at WS 85, and are accessible through access covers located on the wing lower skin. The flap actuators mechanically rotate sectors attached to the flaps through adjustable push-pull tubes.
      (9) The flap position potentiometer (R18) is mounted adjacent to the LH flap sector, between a flap position switch and a flap limit switch.
      (10) The vertical scale flap position indicator, mounted in the center instrument panel, provide the crew with visual indication of flap position. The indicator face consists of a scale, which has markings for UP (0°), 8°, 20°, and DN (40°), and an indicator pointer located on the right margin of the scale. The flap position potentiometer is utilized in sending a flap position signal to the indicator. The indicator operates on 28 vdc supplied through the TAB & FLAP POSN circuit breaker located on the copilot's circuit breaker panel. On *Aircraft 35-675 and Subsequent; 36-064 and Subsequent,* relay (K7104), located on the center pedestal, removes residual voltage to the flap position indicator when emergency power is applied.

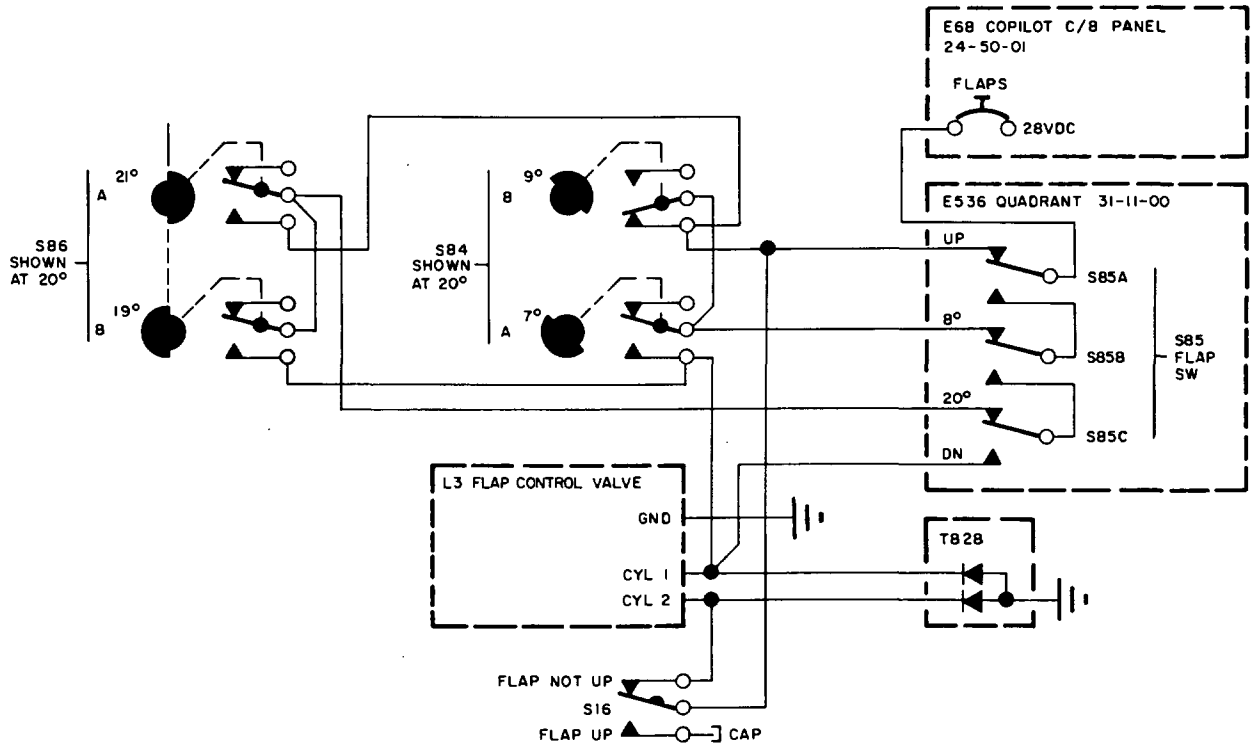
2. **Operation** (See Figures 2, 3, and 4.)
   A. Flap motion is a combination of aft movement to increase wing lift area, and tilting down movement to increase drag. Flap tracks installed at L and RWS 41 and 107 direct flap travel. Maximum flap extension is 40° (+5°/-0°) down. Flaps should travel from full up position (0°) to full down position in ten seconds or less. Flap system operating pressure is 1500 (±50) psi.
   B. The flap control system operates on 28 vdc supplied through the FLAPS circuit breaker located on the copilot's circuit breaker panel.
   C. Setting the Flap Switch to the 8° position directs 28 vdc through the LH limit switch (S84) lower deck (7°) to the flap control valve. The control valve directs hydraulic fluid to the down port of each flap actuator. As the flaps extend below 7°, the limit switch lower deck (7°) opens and de-energizes the flap control valve. With the flap control valve de-energized, the valve returns to neutral and locks the flaps at 8° down. Should a transient aerodynamic load cause the flaps to rise above 7°, the limit switch lower deck (7°) will again energize the down side of the valve and extend the flaps to the 8° position. If a transient aerodynamic load should cause the flaps to extend below 9°, the LH limit switch middle deck (9°) will energize the up side of the control valve and retract the flaps to the 8° position.
   D. Setting the Flap Switch to the 20° position directs 28 vdc through the RH limit switch (S86) lower deck (19°) to the control valve. The control valve directs hydraulic fluid to the down port of each actuator. As the flaps extend below 19°, the limit switch lower deck (19°) opens and de-energizes the control valve. With the flap control valve de-energized, the valve returns to neutral and locks the flaps at 20°. Should a transient aerodynamic load cause the flaps to rise above 19°, the limit switch lower deck (19°) will again energize the control valve and extend the flaps to the 20° position. If a transient aerodynamic load should cause the flaps to extend below 21°, the RH limit switch (S86) middle deck (21°) will energize the up side of the control valve and retract the flaps to the 20° position.
   E. Setting the Flap Switch to the DN (40°) position directs 28 vdc to the down side of the flap control valve. The control valve directs hydraulic fluid to the down port of each actuator. The flaps then extend to 40° (+5°/-0°) at which time the flap actuators bottom out. When the flaps reach full extension, the down pressure will remain, maintaining the flaps in the full extend position.
   F. On *Aircraft 35-002 thru 35-531 and 36-002 thru 36-053 not modified per SB35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference,"* setting the Flap Switch to the UP (0°) position directs 28 vdc through the flap control valve switch (S16) to the flap control valve. The control valve directs hydraulic fluid to the up port of each actuator. When the flaps reach the fully retracted position (0°), the flap control valve switch (S16) de-energizes the control valve (returns to neutral), stopping and locking the flaps in the full retract position.
   G. An aural warning system is electrically connected to the flaps and the landing gear. With the flaps extended beyond the 25° position and the landing gear retracted, the aural warning horn will sound. The aural warning will continue to sound until either the gear is down and locked or the flaps are retracted above 25°.
   H. When the flaps are extended below 25°, the spoilers will work independently of each other and in conjunction with the ailerons to aid in low speed control. During this aileron augmentation mode, the spoilers are controlled by an electrical aileron monitoring system. (Refer to 27-10-00.)

      NOTE: With Spoiler Switch set to EXT position (spoilers extended), the SPOILER annunciator will flash when flaps are extended past 3°.

   I. After extended parking time with power off, the flaps may droop and the spoilers rise. This is normal. Upon starting either the engines or the auxiliary hydraulic pump, the flaps and spoilers will return to the preselect position.

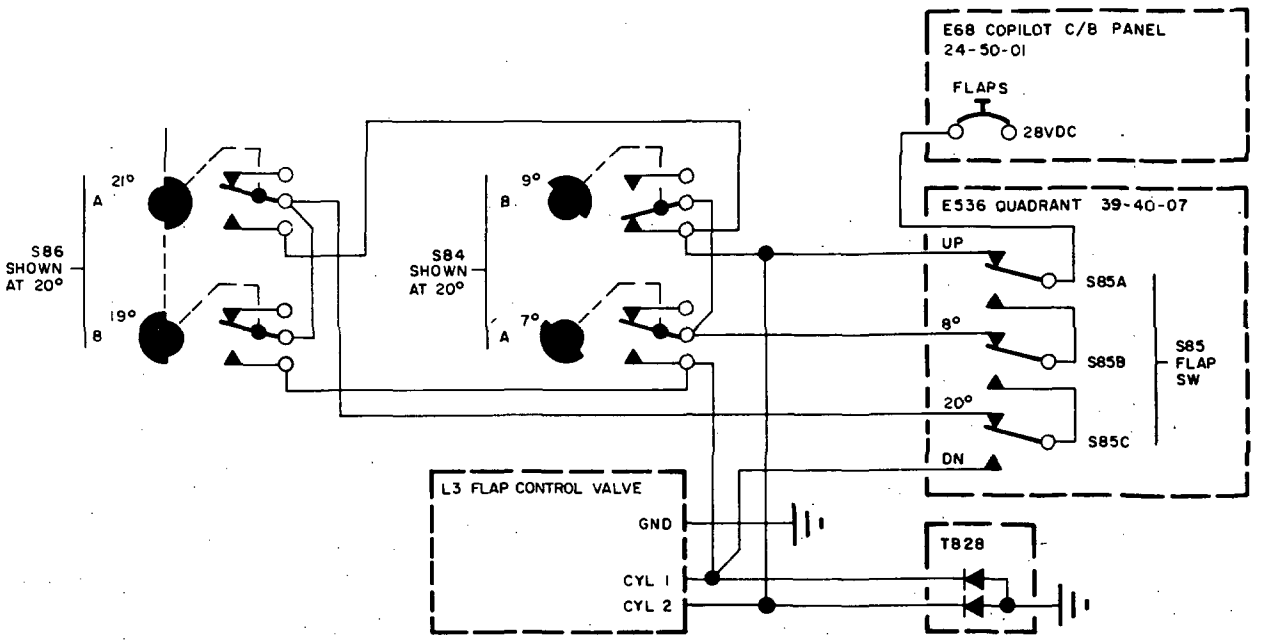


Flap Hydraulic System Components Installation
Figure 1



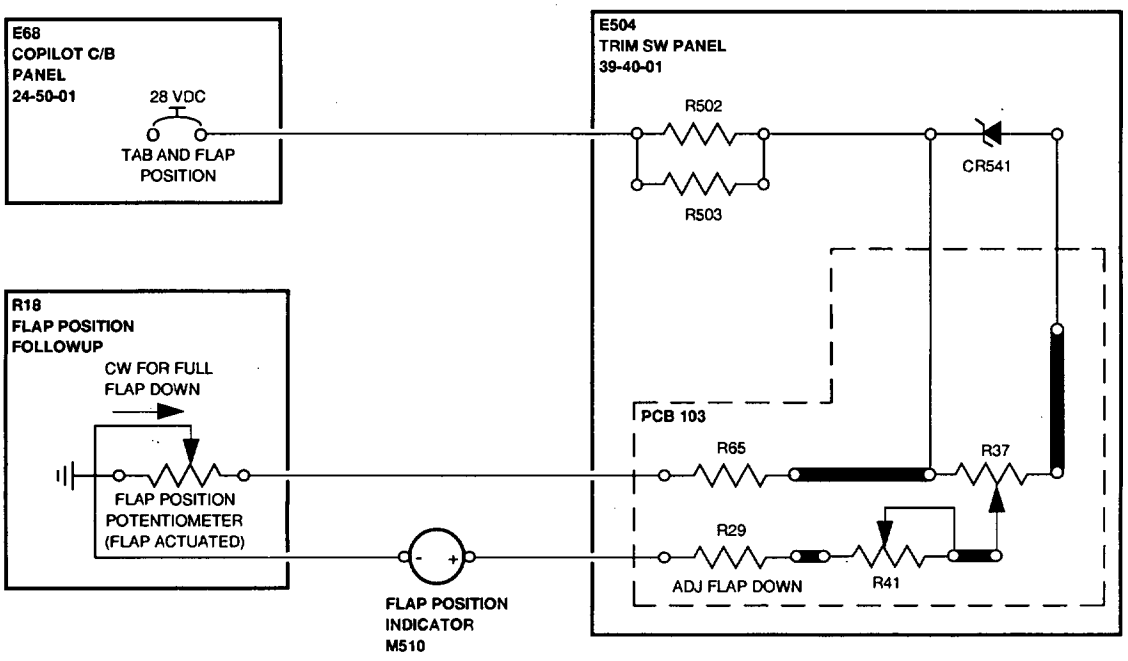
Aircraft 35-417, 35-419, 35-477, 35-479, 35-483 thru 35-531 and 36-051 thru 36-053, except 35-447 and 35-462 not modified per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference."

Flap System Electrical Schematic (Flap Control)
Figure 2 (Sheet 1 of 4)

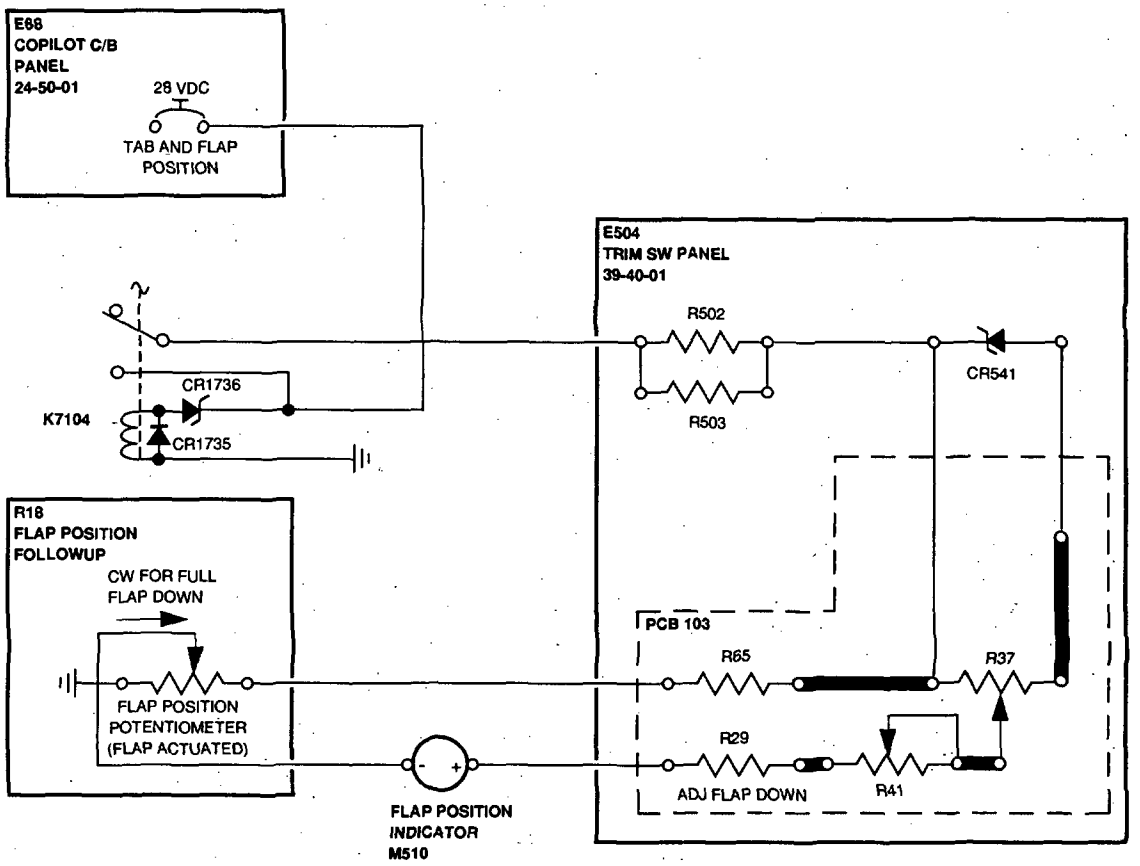


Aircraft 35-532 and Subsequent; 36-054 and Subsequent and prior aircraft modified per SB 35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference."

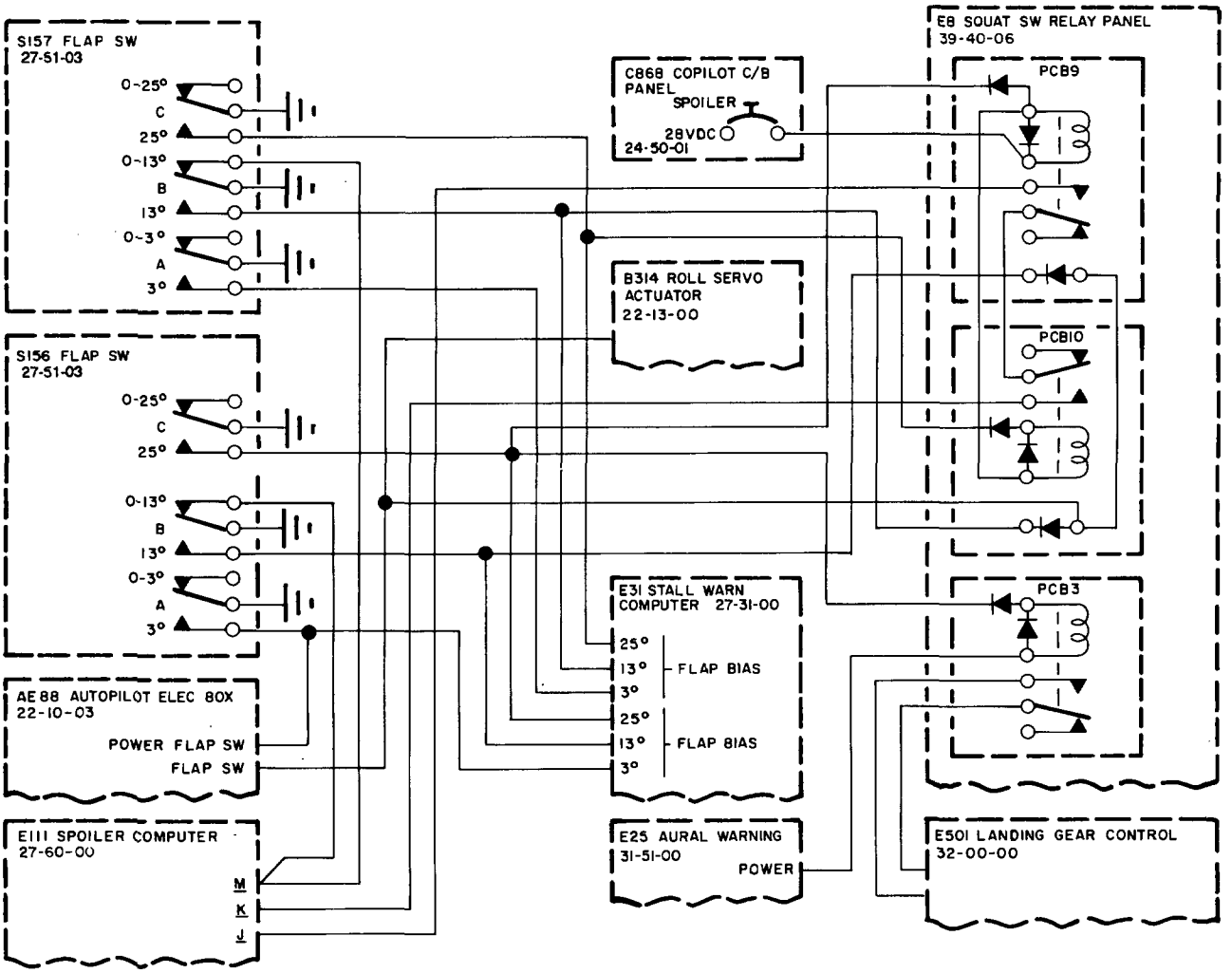
Flap System Electrical Schematic (Flap Control)
Figure 2 (Sheet 2 of 4)



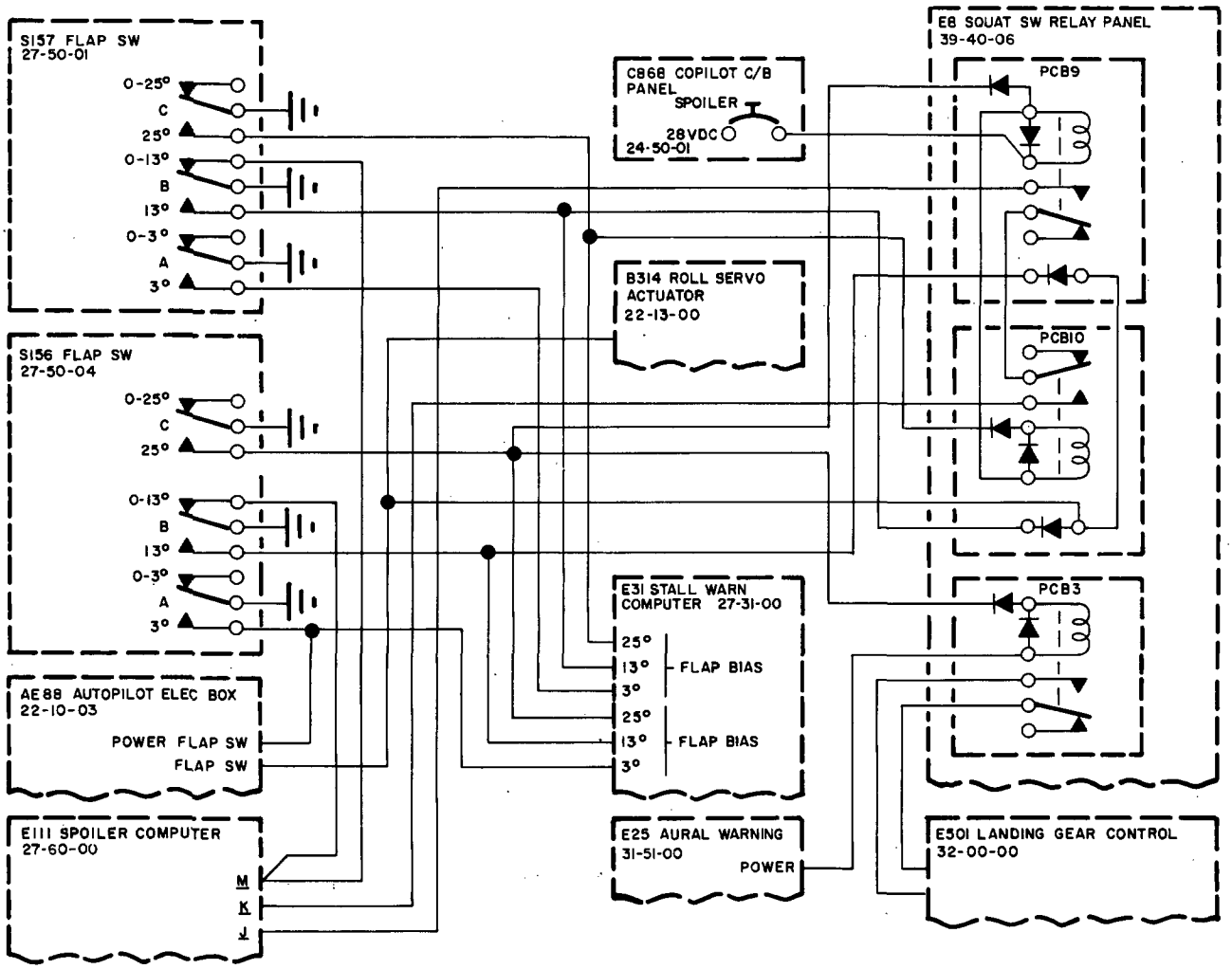
Flap System Electrical Schematic (Position Indicating)
Figure 2 (Sheet 3 of 4)



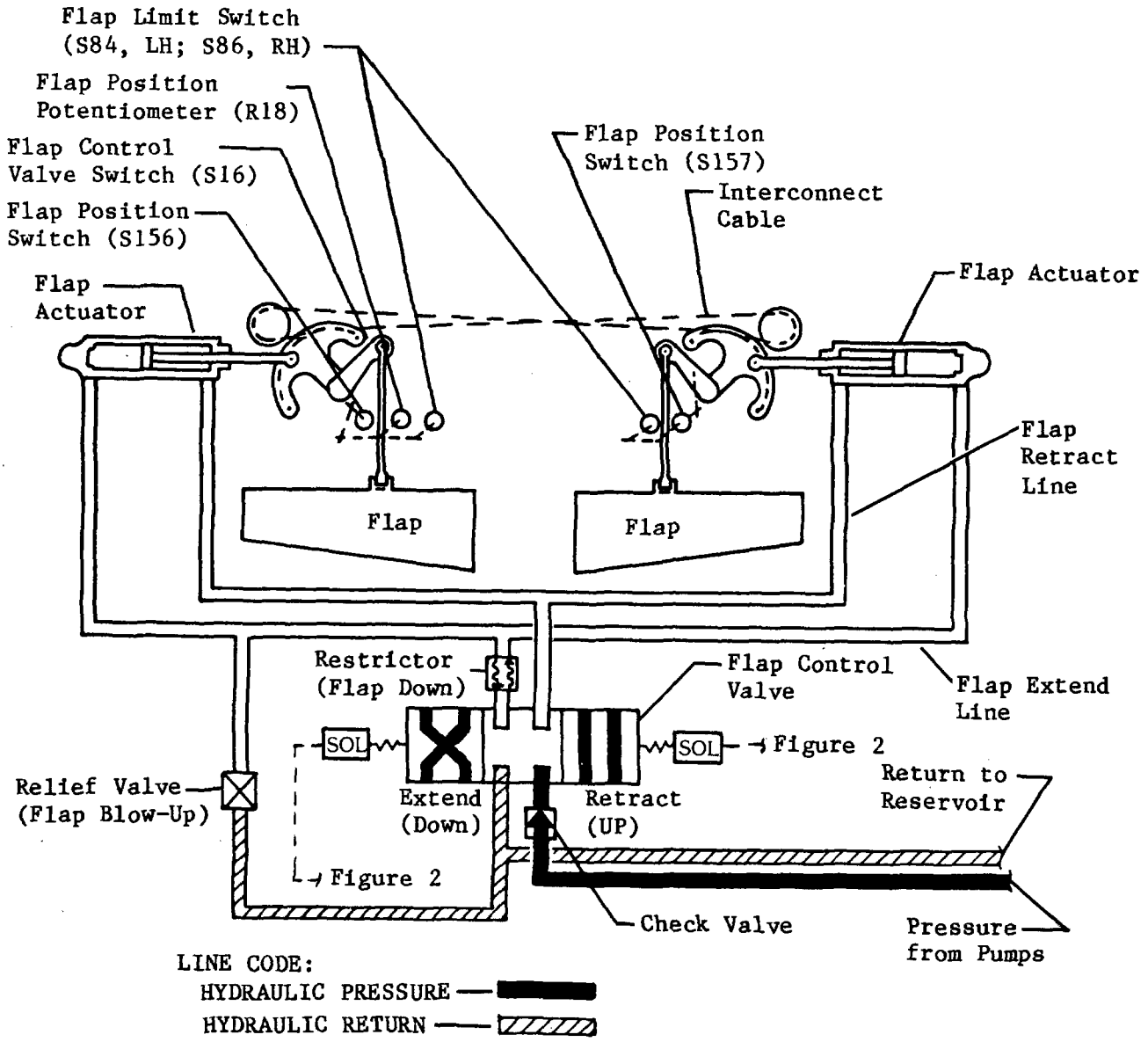
Flap System Electrical Schematic (Position Indicating)
Figure 2 (Sheet 4 of 4)



Flap Aural Warning System
Figure 3 (Sheet 1 of 2)



Flap Aural Warning System
Figure 3 (Sheet 2 of 2)



Flap System Hydraulic Schematic
Figure 4

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Tensiometer		Commercially Available	Adjust cable tension.
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	Measure angles.
Digital Protractor (DP-45) (Alternate)	02475-01	Lucas & Schaevitz Pennsauken, NJ	Measure angles.
Rigging Pin		Local Manufacture	Cable rigging.

2. Adjustment/Test

NOTE: Rigging pin is fabricated from 3/16 inch round steel stock. Make 90° bend, approximately 8-1/2 inches from end of 12 inch piece of stock. Paint short end of piece red (for visibility).

A. Rig Flap System (See Figure 201.)

WARNING: REMAIN CLEAR OF SPOILERS DURING OPERATION TO PREVENT BODILY INJURY.

NOTE: The flaps shall be synchronized to within 2° of each other at all times.

(1) Remove flap sector access covers.
(2) Adjust flap control cable tension in accordance with procedures outlined in step 2.C.
(3) With flaps in retract position, align rigging pin holes in sectors with holes in brackets, and insert rigging pins.
(4) Disconnect actuator rod ends from both left and right sectors.
(5) Check for 0.03 inch minimum clearance between the sectors and wing spar 8.
(6) Adjust flap push-pull tubes until flap rollers contact forward end of flap tracks.
(7) With hydraulic actuators fully retracted (bottomed internally), adjust rod end terminals so that terminals are aligned with holes in sectors where actuators are normally connected.
(8) Turn rod ends 1/2 turn clockwise (shortening retracted length of actuators approximately 0.02 inch), to preload flap track rollers snug into forward end of flap nose track.
(9) Attach rod end terminals to sectors and safety wire terminals.
(10) On *Aircraft not modified per SB35/36-27-15, "Inspect Flap Sector Mounting Brackets for Cracks and Flap System for Interference"*, adjust flap control valve microswitch to actuate with sectors at rigged position.
(11) Remove rigging pins.
(12) Using the Flap Switch on center pedestal, check for proper flap travel. Full down travel should be 40 (+5, -0)°.
(13) With flaps in extended position, spanwise movement (end play) shall be 0.34 inch maximum. If spanwise movement exceeds limits, replace buttons on cam followers. Ensure that there is positive clearance between nose roller and aft end of nose roller track slot (4 places).

(14) Retract flaps to the 20° down position. With all flap end play positioned to the extreme outboard position, clearance between the flap and aileron and the flap and aileron trim tab shall be a minimum of 0.100 inch.
(15) Inspect to ensure that the following items do not occur:
(a) Interference between flap sector and switch mounting bracket.
(b) Contact between flap and spoiler actuator bracket.
(c) Contact between flap and switch actuating arm on spoiler.
(d) Contact between flap and lower wing surface. Make any necessary adjustments.
(16) Install access covers.
(17) Perform operational checks of Stall Warning System (27-31-00), Spoiler Warning System (27-60-00), and the Landing Gear Aural Warning System (Chapter 32); make adjustments as required.

B. Perform Gated Flaps System Functional Test

WARNING: REMAIN CLEAR OF SPOILERS DURING OPERATION TO PREVENT BODILY INJURY.

CAUTION: ENSURE THAT THE FLAPS AND ALL SWITCHES ARE RIGGED PRIOR TO PERFORMING THIS FUNCTIONAL TEST.

(1) Connect electrical ground power unit to aircraft.
(2) Set Battery Switches to BAT 1 and BAT 2.

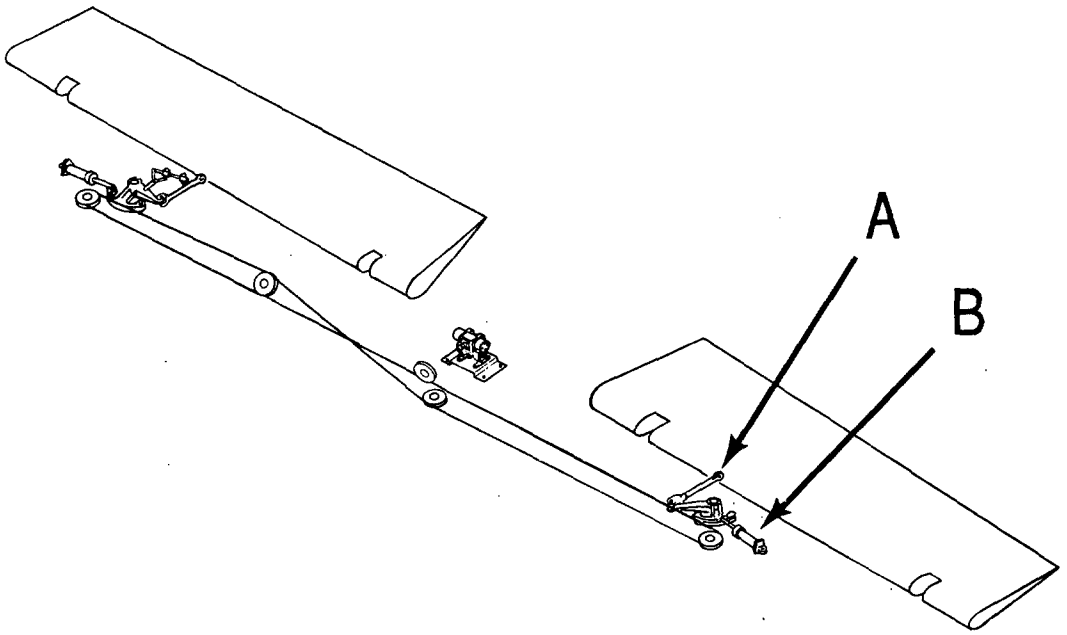
NOTE: Ensure that flaps are completely retracted.

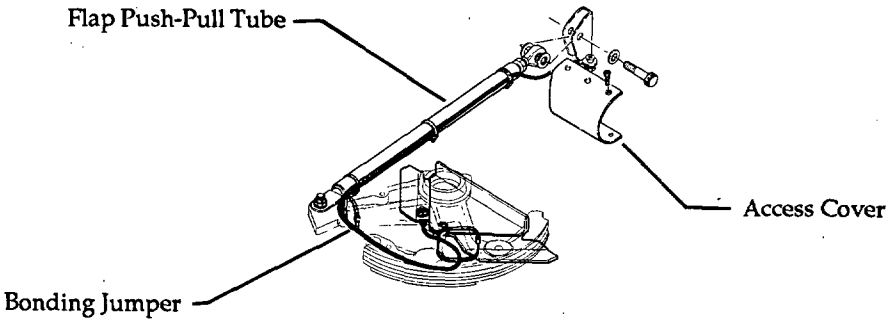
(3) Install clinometer on flaps.
(4) Set Flap Control Switch to 8° position. Flaps shall travel to 8 (±0.5)° position and stop.
(5) Set Flap Control Switch to 20° position. Flaps shall travel to 20 (±0.5)° position and stop.
(6) Set Flap Control Switch to DN position. Flaps shall travel to 40 (+5; -0)° position and stop.
(7) Set Flap Control Switch to 20° position. Flaps shall travel to 20 (±0.5)° position and stop.
(8) Set Flap Control Switch to 8° position. Flaps shall travel to 8 (±0.5)° position and stop.
(9) Set Flap Control Switch to UP position. Flaps shall travel to full up position (0°).
(10) If the gated flap system does not operate as described in steps B.(4) thru B.(9), rig system and repeat steps B.(1) thru B.(9).
(11) Set Battery Switches to OFF.
(12) Disconnect electrical ground power unit from aircraft.
(13) Remove clinometer from flaps.

C. Adjust Flap Control Cable Tension

NOTE:
- Prior to flap cable tension adjustment, torque wrench tensiometer adapter calibration must be accomplished. (Refer to Adjustment/Test, 27-00-00.)
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (24°C). The Cable Tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to Adjustment/Test, 27-00-00.)

(1) With flaps in retract position, remove flap sector access cover, align rigging pin holes in sector with holes in bracket, and insert 3/16" rigging pin.
(2) Remove wing access panel, located on spar 7 in the left wheel well, to gain access to flap control cables and turnbuckles.
(3) Remove locking clips from turnbuckles.

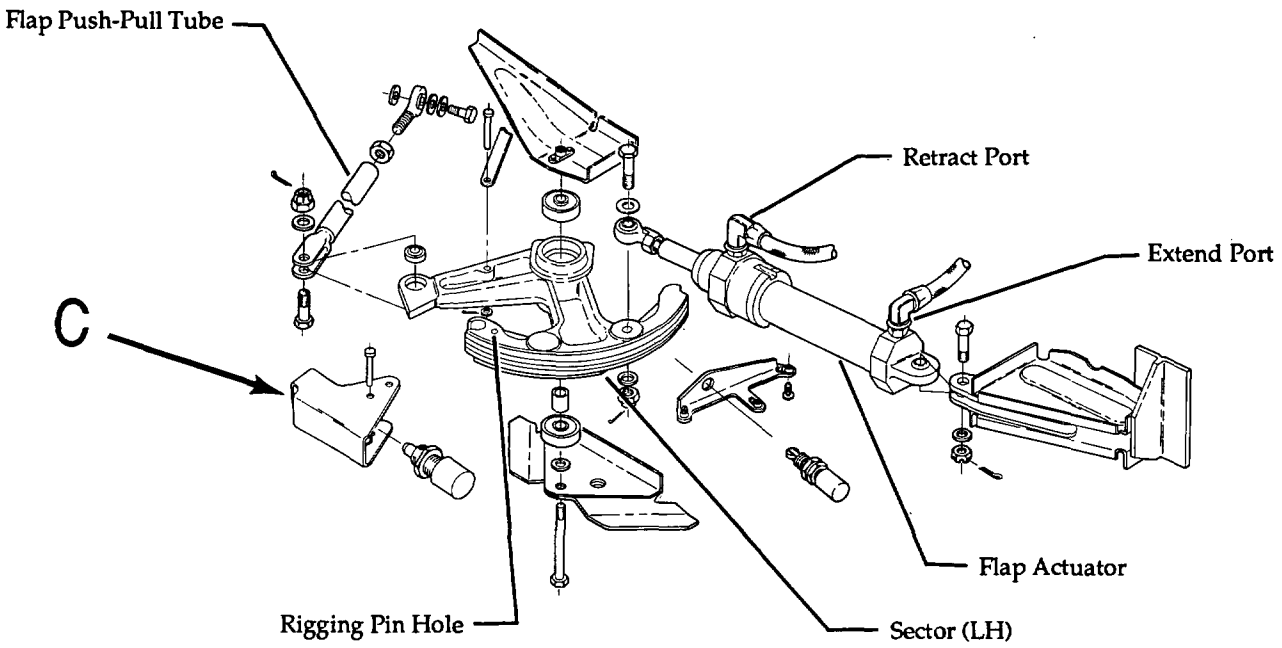




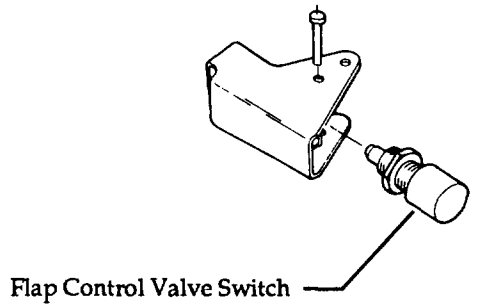
Detail A

8-50B
A8-6C-6

Flap System Rigging
Figure 201 (Sheet 1 of 2)



Detail B



Applicable Only To Aircraft Not Modified Per SB35/36-27-15, "Inspect Flap Sector Mounting Brackets For Cracks and Flap System For Interference".

Detail C

B8-6C-2

Flap System Rigging
Figure 201 (Sheet 2 of 2)

(4) If a new cable is being installed, perform the following:
  (a) Adjust flap cable tension to 200 (±10) pounds (90.7 [±4.5] kg).
  (b) Cycle flaps 25 times to seat in and pre-stretch new cable.
  (c) Reduce cable tension.
(5) Adjust flap control cable tension to 100 (±25) pounds (45.4 [±11.3] kg).

NOTE:
- The threads at opposite ends of the turnbuckle barrel shall be started at the same time so thread engagement on both ends will be approximately equal.
- The final adjusted turnbuckle combination shall have no more than three threads of the terminal, fork, or eye exposed outside the barrel of the turnbuckle.

(6) Once proper cable tension has been obtained, remove rigging pin and operationally check flap control system. (Refer to Inspection/Check, this section.)
(7) Install locking clips in turnbuckles.
(8) Install all previously removed access panels.
(9) Restore aircraft to normal.

3. Inspection/Check

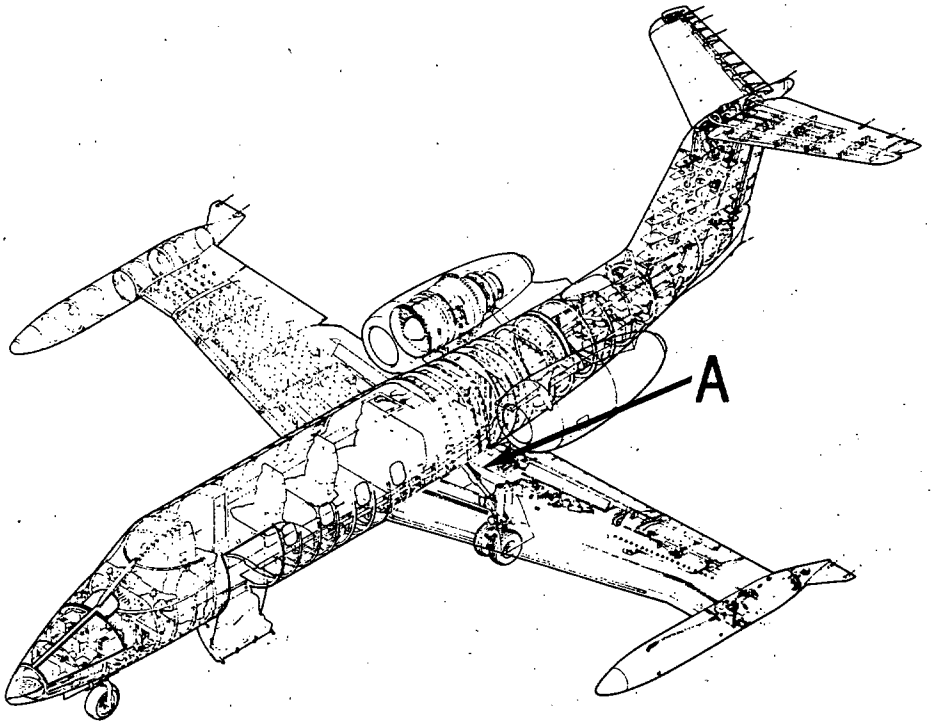
A. Perform Flap Control Cable Tension Check (See Figure 202.)

NOTE:
- Perform flap control cable tension check in accordance with the current inspection intervals specified in Chapter 5.
- Ambient temperature has a direct affect on cable tension. All cable tensions specified in the following procedures are with ambient temperature at 75°F (23.9°C). The cable tension Temperature-Load Correction chart shall be utilized to determine proper tensioning values. (Refer to 27-00-00, Adjustment/Test.)

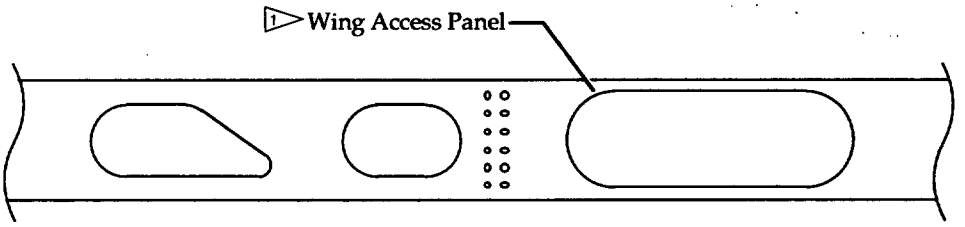
(1) Remove wing access panel, located on spar 7 in the left wheel well, to gain access to flap control cables and turnbuckles.
(2) Attach tensiometer to flap control cables. Flap cable tension shall be 100 (±25) pounds at 75°F (45.4 [±11.3] kg at 24°C).

NOTE:
- The flap cable must be lying against the back of the tensiometer when pressure is applied so that no extra pull is applied to the cable.
- Avoid pushing or pulling on tensiometer or cable while a reading is being taken.

(3) Rig flap system if readings do not fall within tolerances. (Refer to Adjustment/Check, this section.)
(4) Remove tensiometer from flap control cable and install wing access panel.
(5) Restore aircraft to normal.



1 Flap control cables and turnbuckles located behind wing access panel.



(VIEW LOOKING AFT AT SPAR 7, LH WHEEL WELL)

Detail A

Flap Control Cable Tension Check
Figure 202

1. Removal/Installation

NOTE:
- Removal and installation of RH and LH flaps are identical.
- Two installation procedures are provided, one for installing existing flaps and one for installing new flaps. When installing a new flap it is necessary to trim the flap fairing to proper fit.

A. Tools and Equipment

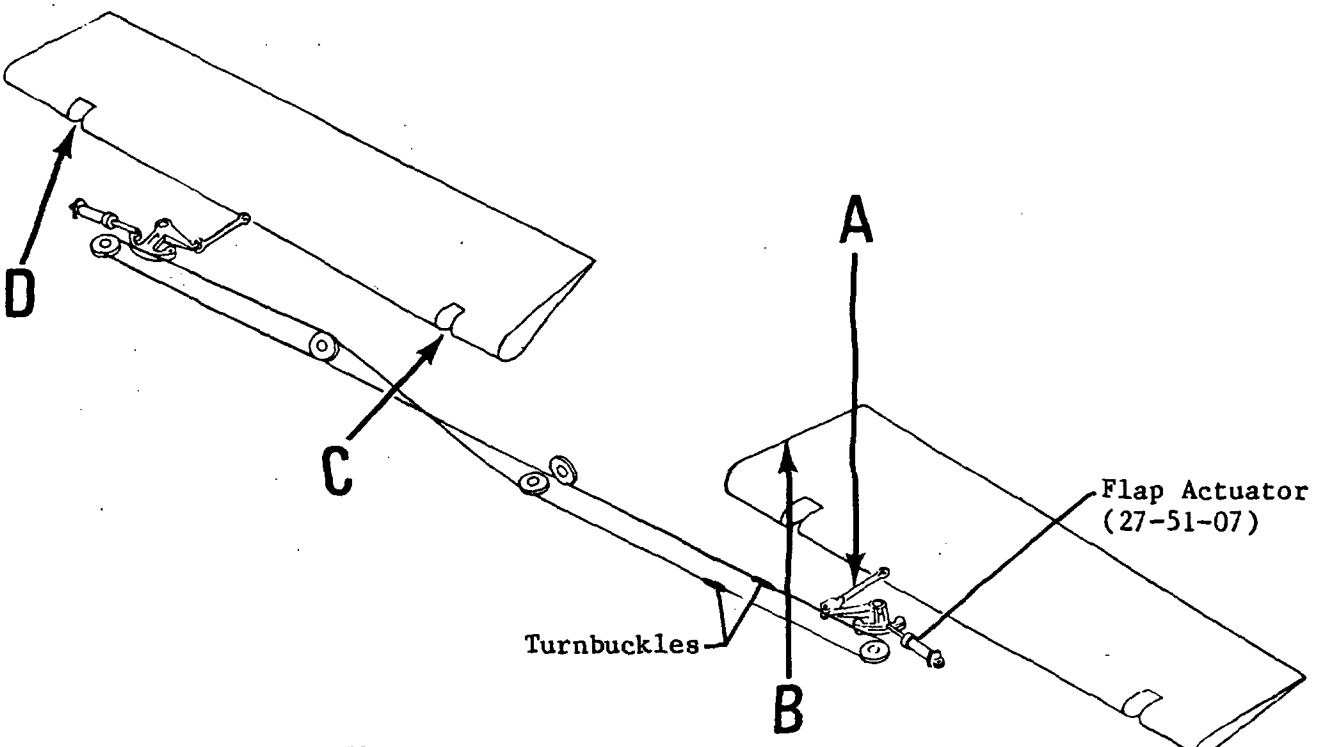
NOTE: Equivalent substitutes may be used in lieu of the following:

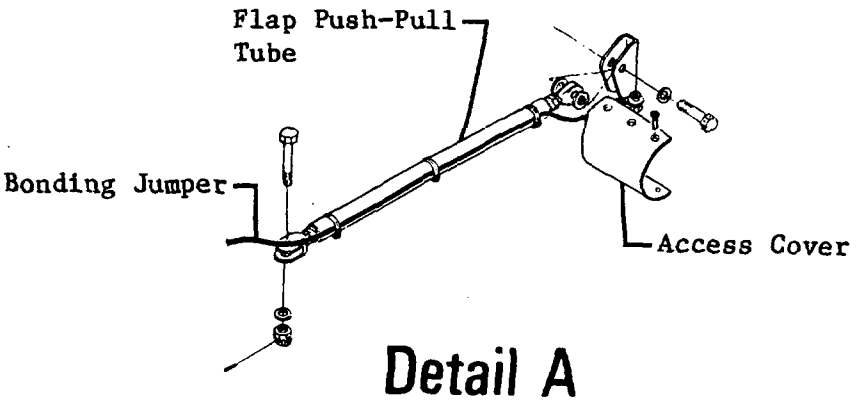
NAME	PART NUMBER	MANUFACTURER	USE
Grease	MIL-G-81322	Commercially Available	Lubrication.
Zerk Gun		Commercially Available	Apply grease.
90° Rubber Tip	327265	Alemite	Apply grease.
Rigging Pin		Local Manufacture	Rigging.
Clean, Cotton Cloth		Commercially Available	Cleaning.
Protective Tape	#8681	3M Company St. Paul, MN	Prevent wing/ flap chafe condition.
Protective Tape	#8641	3M Company St. Paul, MN	Prevent wing/ flap chafe condition.
Isopropyl Alcohol		Commercially Available	Clean surface.
Adhesion Promoter	#86	3M Company St. Paul, MN	Prepare surface for tape.

B. Remove Flap (See Figure 201.)

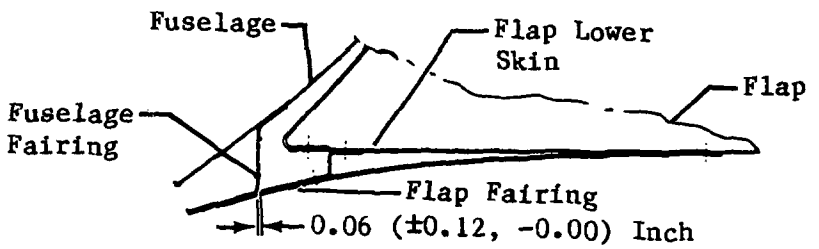
WARNING: SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM THE AIRCRAFT WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELECTRICAL POWER.

(1) With hydraulic and electrical power supplied to aircraft, fully extend spoilers and flaps.
(2) Remove hydraulic power from aircraft.





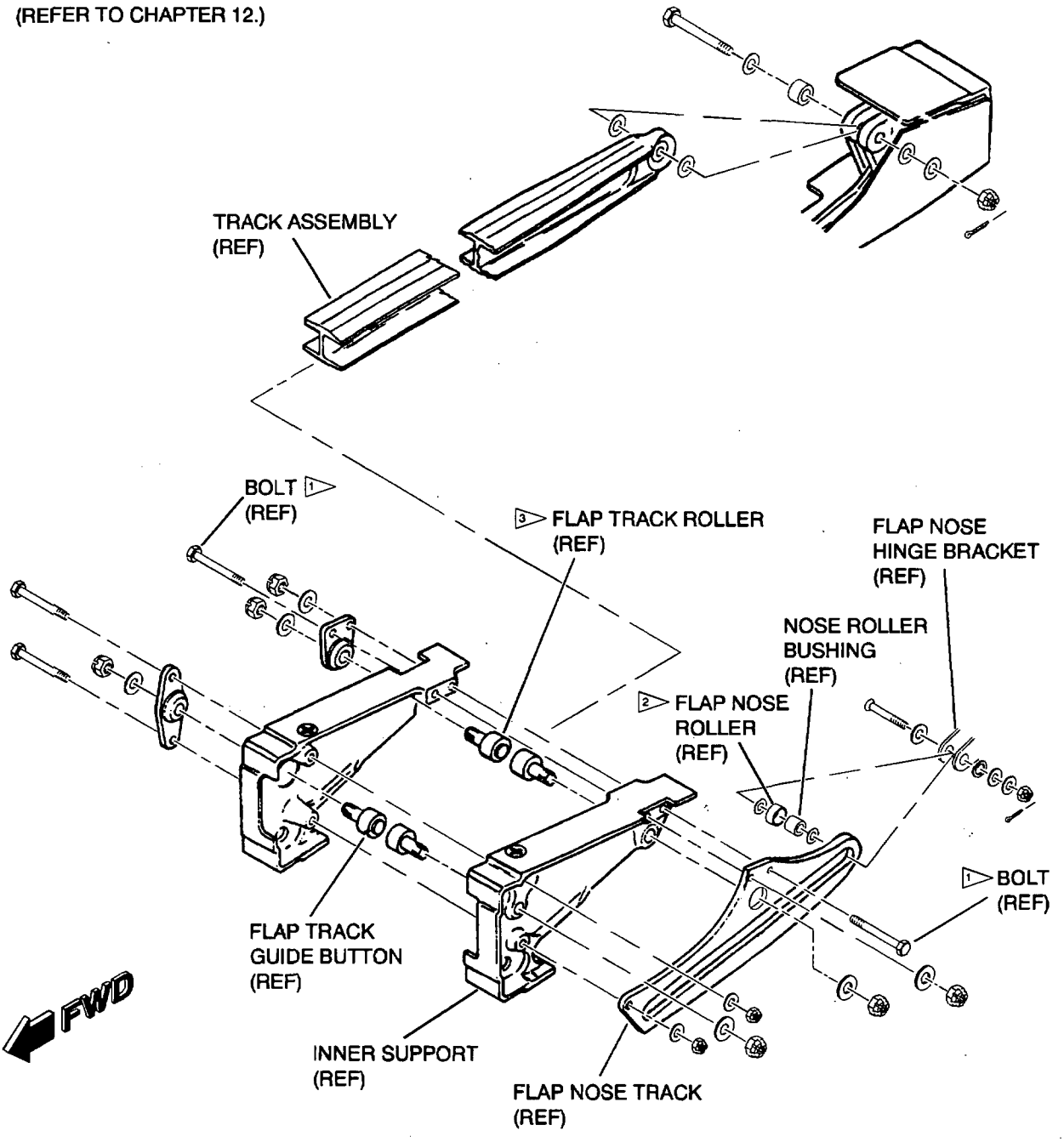
Detail A



Detail B

Flap System Installation
Figure 201 (Sheet 1 of 2)

1 BE SURE HEADS OF THESE BOLTS ARE LOCATED AS SHOWN. IF NOT INSTALLED AS SHOWN, BOLTS WILL NOT CLEAR WHEN FLAPS ARE RETRACTED.
2 DO NOT LUBRICATE THE FLAP NOSE ROLLERS.
3 LUBRICATE FLAP TRACK ROLLERS. (REFER TO CHAPTER 12.)

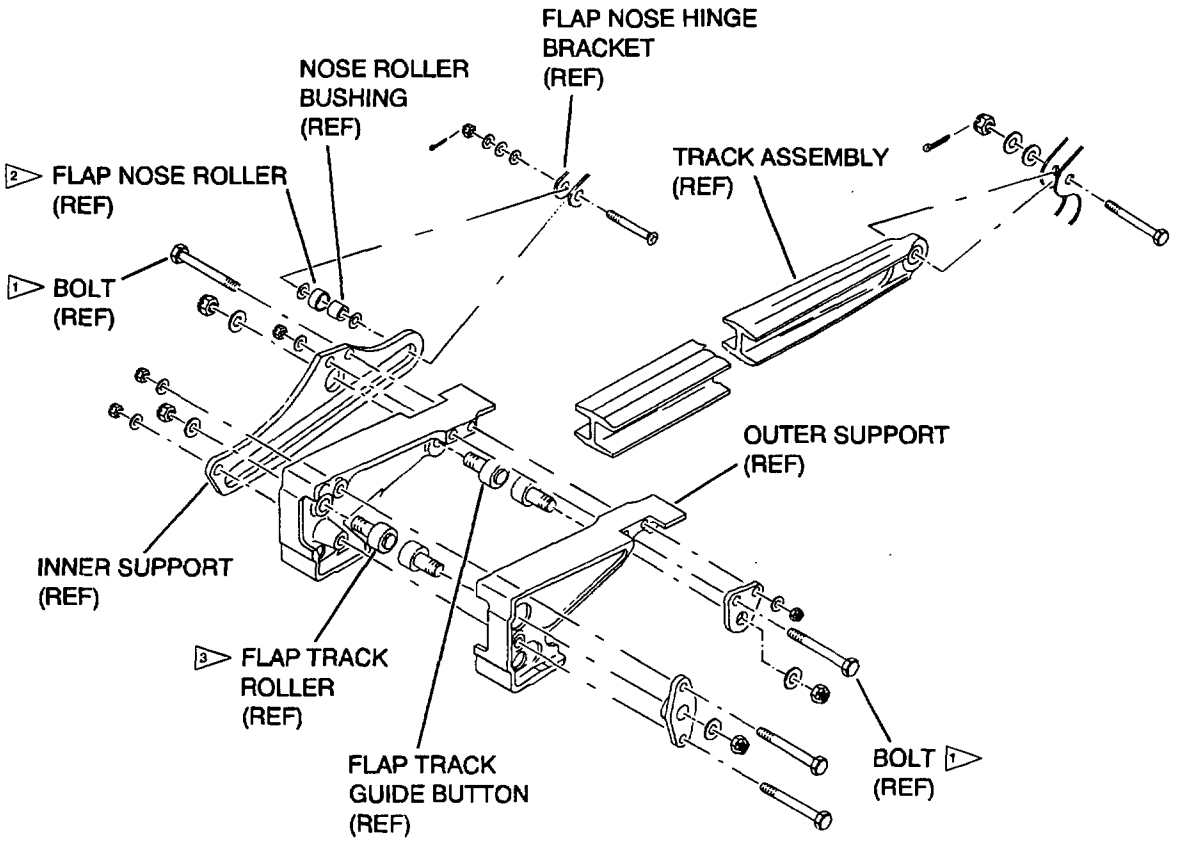


(INBOARD TRACK ASSEMBLY)

Detail C

M35-275101-201-02

Flap System Installation
Figure 201 (Sheet 2 of 3)



(OUTBOARD TRACK ASSEMBLY)

Detail D

M35-275101-201-03

Flap System Installation
Figure 201 (Sheet 2 of 3)

(3) Remove electrical power from aircraft.

NOTE: Rigging pin is fabricated from 3/16 inch round steel stock. Make 90° bend, approximately 8-1/2 inches from end of 12 inch piece of stock. Paint short end of piece red (for visibility).

(4) Remove flap sector access cover from underside of wing and insert rigging pin in sector.
(5) Disconnect bonding jumper (when installed) from flap assembly. A bonding jumper is located adjacent to the outboard and the inboard track assemblies.
(6) Remove bolts securing flap to flap nose track releasing rollers, nose roller bushings, and washers from flap nose track.
(7) Remove flap push-pull access cover and disconnect bonding jumper and push-pull tube.
(8) Remove flap from aircraft.

C. Installation of Existing Flap (See Figure 201.)
(1) Prior to installation of new flap assembly, check all components.
(a) Check all cam followers for scoring and excessive wear. Replace as necessary.
(b) Check all flap tracks for rust.
(c) Check pushrod assembly end bearings. Replace as necessary.
(d) Check bearings in flap tracks. Replace as necessary.
(e) Check flap track guide buttons, located on cam followers, for excessive wear. Replace as necessary.
(f) *On Aircraft 35-002 thru 35-676 and 36-002 thru 36-063 not modified per SB 35/36-27-32*, inspect the hole in the flap sector support brackets for cracks and elongation.
(2) Clean flap tracks and flap nose tracks.
(3) Lubricate cam followers with MIL-G-81322 grease using a zerk gun and a 90° rubber tip adapter. (Refer to Chapter 12.)
(4) Align flap and flap tracks with cam followers and push-pull tube with push-pull tube access hole in wing rear spar.
(5) Insert flap tracks on cam followers and push-pull tube into access hole, ensuring that flap tracks align on forward rollers and flap nose track is between flap nose hinge brackets.
(6) With flap nose tracks on flap nose hinge brackets, install flap nose rollers, nose roller bushings, and washers using attaching parts.
(7) Connect flap bonding jumper at rear wing spar and push-pull tube to flap sector.
(8) Attach flap push-pull tube to sector by inserting bolt from the bottom side and secure with nut, washer, and cotter pin. Remove rigging pin from aft sector.
(9) Rig flaps. (Refer to 27-50-00, Adjustment/Test.)
(10) If required, replace protective tape on lower wing skin and upper wing skin lower surface.
(11) Install sector assembly access cover.
(12) Restore hydraulic power to aircraft.
(13) Restore electrical power to aircraft.
(14) Stow flaps and spoilers in full retract position.
(15) Restore aircraft to normal.

D. Installation of Replacement Flap (See Figure 201.)

NOTE: Attaching hardware not supplied with a new flap should be removed from the old flap and used on the new flap.

(1) Prior to installation of new flap assembly, check all components.
(a) Check all cam followers, flap nose rollers, and flap nose roller bushings for scoring and wear. Replace as necessary.
(b) Check all flap tracks for rust.

(c) Check pushrod assembly end bearings. Replace as necessary.
(d) Check bearings in flap tracks. Replace as necessary.
(e) Check flap track guide buttons, located on cam followers, for wear. Replace as necessary.
(f) *On Aircraft 35-002 thru 35-676 and 36-002 thru 36-063 not modified per SB 35/36-27-32*, inspect the hole in the flap sector support brackets for cracks and elongation.

(2) Clean flap tracks and flap nose tracks.
(3) Lubricate cam followers with MIL-G-81322 grease using zerk gun with a 90° rubber tip adapter. (Refer to Chapter 12.)
(4) Install flap tracks on flaps using attaching parts removed from old flap.
(5) Connect bonding jumper and push-pull tube to flap.
(6) Align flap and flap tracks with cam followers and push-pull tube with push-pull tube access hole in wing rear spar.
(7) Insert flap tracks on cam followers and push-pull tube into access hole, ensuring that flap tracks align on forward cam followers and flap nose track is between flap nose hinge brackets.
(8) Raise flap by hand to retracted position.

NOTE: Flap cannot be raised to the full retract position until after the flap fairing has been trimmed.

(9) Measure and mark flap fairing so that it may be trimmed to the gap shown in Figure 201, Detail B.
(10) Lower flap and trim along the trim line previously established.
(11) Raise flap by hand to full retract position and check for proper gap between fuselage fairing and flap fairing. Trim as necessary to obtain uniform gap.
(12) After gap has been set on flap assembly, apply protective tape #8641 to lower wing skin as follows:
(a) Clean any oxides off lower wing skin from trailing edge forward 1 inch with fine abrasive pad. Touch up chemical film. (Refer to Chapter 12.)
(b) Clean surface with a 50/50 mixture of water and isopropyl alcohol.
(c) After surface is dry, apply a light coat of 3M #86 adhesion promoter with a clean, lint-free cloth. Allow promoter to dry 15 to 20 minutes.
(d) Install #8641 tape to lower wing surface aligning edge of tape with trailing edge of wing skin.
(e) Use a roller to work out any bubbles being careful not to stretch the tape.
(f) Clean any oxides off upper wing skin lower surface from trailing edge forward 1-1/2 inches with fine abrasive pad. Touch up chemical film. (Refer to Chapter 12.)
(g) Clean surface with a 50/50 mixture of water and isopropyl alcohol.
(h) After surface is dry, apply a light coat of 3M #86 adhesion promoter with a clean, lint-free cloth. Allow promoter to dry 15 to 20 minutes.
(i) Install #8681 tape to lower wing surface aligning edge of tape with trailing edge of wing skin.
(j) Use a roller to work out any bubbles being careful not to stretch the tape.
(13) With flap nose tracks on flap nose hinge brackets, install flap nose rollers, nose roller bushings, and washers using attaching parts.
(14) Connect flap bonding jumper to rear wing spar.
(15) Attach flap push-pull tube to sector by inserting bolt from bottom side and secure with nut, washer, and cotter pin.
(16) Rig flaps. (Refer to 27-51-00, Adjustment/Test.)
(17) Perform functional test of stall warning system (27-31-00, Adjustment/Test), operational check of spoiler warning system (27-60-00, Inspection/Check), and landing gear aural warning system (Chapter 32). Make adjustments as required.

PRESELECT ASSEMBLY AND PRESELECT SWITCH - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

A. Remove Preselect Assembly (See figure 201.)

(1) Remove electrical power from aircraft.

(2) Gain access to flap preselect assembly through forward pedestal access panels.

CAUTION: THE FLAP PRESELECT ASSEMBLY IS HELD TOGETHER BY THE KNOB, SPACER, AND SCREWS. TO KEEP THE ASSEMBLY FROM FALLING APART, THE SIDES OF THE LEVER WITH THE SPACER BETWEEN THEM MUST BE SAFETY WIRED TOGETHER AS SOON AS THE HANDLE IS REMOVED. MOVEMENT OF THE LEVER SIDES PRIOR TO SAFETY WIRING LIKELY WILL RELEASE INTERNAL COMPONENTS SUCH AS THE PIN AND ROLLER IN THE CAM SLOT.

(3) Remove attaching parts and flap preselect knob from preselect assembly and safety wire lever sides and spacer together.

(4) Remove attaching parts and preselect assembly link from preselect switch arm.

(5) Remove attaching parts and flap preselect assembly from aircraft.

B. Install Preselect Assembly (See figure 201.)

(1) Position preselect assembly and and secure with attaching parts at swivel points.

(2) Install preselect assembly link to preselect switch arm and secure with attaching parts.

(3) Carefully remove safety wire from aft end of assembly and install flap preselect knob without disturbing relationship of lever sides. Secure with attaching parts.

(4) Install access panels in forward pedestal.

(5) Restore electrical power to aircraft.

C. Remove Preselect Switch (See figure 201.)

(1) Remove electrical power from aircraft.

(2) Gain access to preselect switch through forward pedestal access panels.

(3) Remove cotter pin from preselect switch arm pin and pin.

(4) Disconnect electrical wires from preselect switch.

(5) Remove preselect switch mount bracket screws.

(6) Remove preselect switch assembly, mount bracket, and arm as one unit.

D. Install Preselect Switch (See figure 201.)

(1) Position preselect switch assembly and install mount bracket screws. Do not tighten.

(2) Install preselect switch arm pin and cotter pin.

(3) Connect electrical wires to preselect switch.

(4) Tighten mount bracket screws.

(5) Perform rigging check of preselect switch assembly. (Refer to Adjustment/Test.)

(6) Restore electrical power to aircraft.

(7) Perform gated flap functional test. (Refer to 27-51-00.)

(8) Install access panels on forward pedestal.

2. AJUSTMENT/TEST

A. Tools and Equipment

NAME	NUMBER	MANUFACTURER	USE
Volt-Ohmmeter	Simpson 260 or equivalent		To check for system continuity
Putty	EC1252 or equivalent	Minnesota Mining & Mfg. Co. 3M Center, St. Paul, MN 55144	To torque stripe screws
Protractor Sealant	Pro-Seal 890B2 or equivalent	Coast Pro-Seal & Mfg. Co.	Electrical Sealant

B. Rig Flap Preselect Switch Assembly (See figure 202.)

NOTE:
- ° The flap preselect switch assembly is factory preset to actuate midway between flap positions and should not require any further adjustments. If adjustments are required, only minor adjustments are authorized.
- ° Refer to Removal/Installation procedures to gain access to flap preselect switch.
- ° Ensure flap system is properly rigged prior to installing and adjusting preselect switch assembly.
- ° Adjustment screw "A" will be used for the majority of adjustments unless the deck is totally out-of-rig, then adjustment screw "B" will also be used.
- ° Adjustment screws ("A" or "B") can be rotated clockwise (cw) or counterclockwise (ccw) to make adjustments. During the adjustment of each deck switch, direction of rotation of the adjustment screw cannot be specified except to rotate adjustment screw to obtain the desired setting. Normally the direction will be cw for adjustment screw "A".
- ° Each switch deck has internal stops for the adjustment screws ("A" or "B"). When the travel limit is reached and deck switch does not respond to desired setting, the other adjustment screw (usually "B") will have to be rotated in opposite direction to bring deck switch into operating range.

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(1) Check all three switch decks as follows:
  (a) Set flap preselect handle to UP position.
  (b) Check continuity between terminals NO and C on each deck.
  (c) All switches shall be open in this position.

(2) If initial adjustment of the switches is required Aircraft not equipped with preset flap preselect switch assembly), refer to paragraph C, Initial Rig of Flap Preselect Switch Assembly.

(3) Check Deck "A" switch as follows:
  (a) Set flap preselect handle to 8° position.
  (b) Check continuity between terminals NO and C on Deck "A".
  (c) Switch shall be closed in this position.
  (d) Adjust switch to close when flap preselect handle is approximately midway between UP and 8° positions, if adjustment is required.

  NOTE: Adjustment screw "A" rotation cw moves switch actuation cw. Adjustment screw "A" rotation ccw moves switch actuation ccw. Same action occurs for adjustment screw "B".

  (e) Loosen lock screw on adjustment screw "A" and rotate adjustment screw until Deck "A" switch closes.
  (f) Tighten lock screw on adjustment screw "A".
  (g) More flap preselect handle to 20° and 40° positions, ensure Deck "A" switch remains closed.

(4) Check Deck "B" switch as follows:
  (a) Set flap preselect handle to 20° position.
  (b) Check continuity between terminals NO and C on Deck "B".
  (c) Switch shall be closed in this position.
  (d) Adjust switch to close when flap preselect handle is approximately midway between 8° to 20° positions, if adjustment is required.

  NOTE: Adjustment screw "A" rotation cw moves switch actuation cw. Adjustment screw "A" rotation ccw moves switch actuation ccw. Same action occurs for adjustment screw "B".

  (e) Loosen lock screw on adjustment screw "A" and rotate adjustment screw until Deck "B" switch closes.
  (f) Tighten lock screw on adjustment screw "A".
  (g) Move flap preselect handle to 40° position, ensure Deck "B" switch remains closed.

(5) Check Deck "C" switch as follows:
  (a) Set flap preselect handle to 40° position.
  (b) Check continuity between terminals NO and C on Deck "C".
  (c) Switch shall be closed in this position.
  (d) Adjust switch to close when flap preselect handle is approximately midway between 20° and 40° positions, if adjustment is required.

  NOTE: Adjustment screw "A" rotation cw moves switch actuation cw. Adjustment screw "A" rotation ccw moves switch actuation ccw. Same action occurs for adjustment screw "B".

  (e) Loosen lock screw on adjustment screw "A" and rotate adjustment screw until Deck "C" switch closes.
  (f) Tighten lock screw on adjustment screw "A".

(6) Check all lock screws are tight after adjustments are completed.
(7) Seal all adjustment screws and lock screws with EC1252 putty.
(8) Seal switch terminals with Pro-Seal 890B2.
(9) After adjustment/test is completed, proceed with 1.D.(6) thru 1.D.(8).

C. Initial Rig of Flap Preselect Switch Assembly (See figure 202.)

NOTE: ° Before making initial rig of flap preselect switch assembly, it is suggested that an index mark be made on the mount bracket at the 45° position.
° After securing switch arm to shaft, do not make any further adjustments of switch arm on shaft.

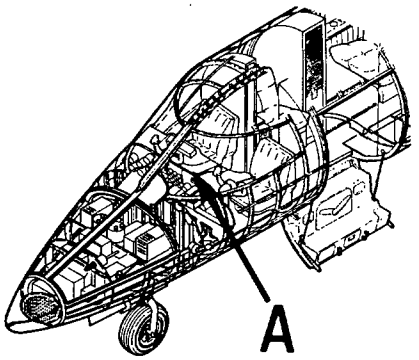
(1) Position flap preselect switch assembly on mount bracket with terminals at 45° mark.
(2) Align index mark on shaft end and switch arm with 45° mark. Secure switch arm with attaching parts.
(3) Secure flap preselect switch and mount bracket assembly and a protractor to bench for ease of making adjustments.
(4) Adjust Deck "A" switch as follows:
  (a) Connect VOM between terminals NO and C
  (b) Set switch arm at 77° on protractor.
  (c) Loosen lock screw on adjustment screw "A" and rotate adjustment screw cw or ccw until switch closes.
  (d) Rotate switch arm toward 0° and check switch opens. This verifies that internal wiper has been positioned correctly and is not 180° out-of-rig.
  (e) Set switch arm at 180° on protractor.
  (f) Loosen lock screw on adjustment screw "B" and rotate adjustment screw cw or ccw until switch opens.

(g) Rotate switch arm ccw to 0° on protractor.
(h) Rotate switch arm cw from 0° to 270° and ccw back to 0°.
(i) Check that switch closes at 77° ±2° and opens at 180° ±10°.
(j) The switch must be closed between 77° to 180° travel range.
(k) Tighten lock screws on adjustment screws "A" and "B".

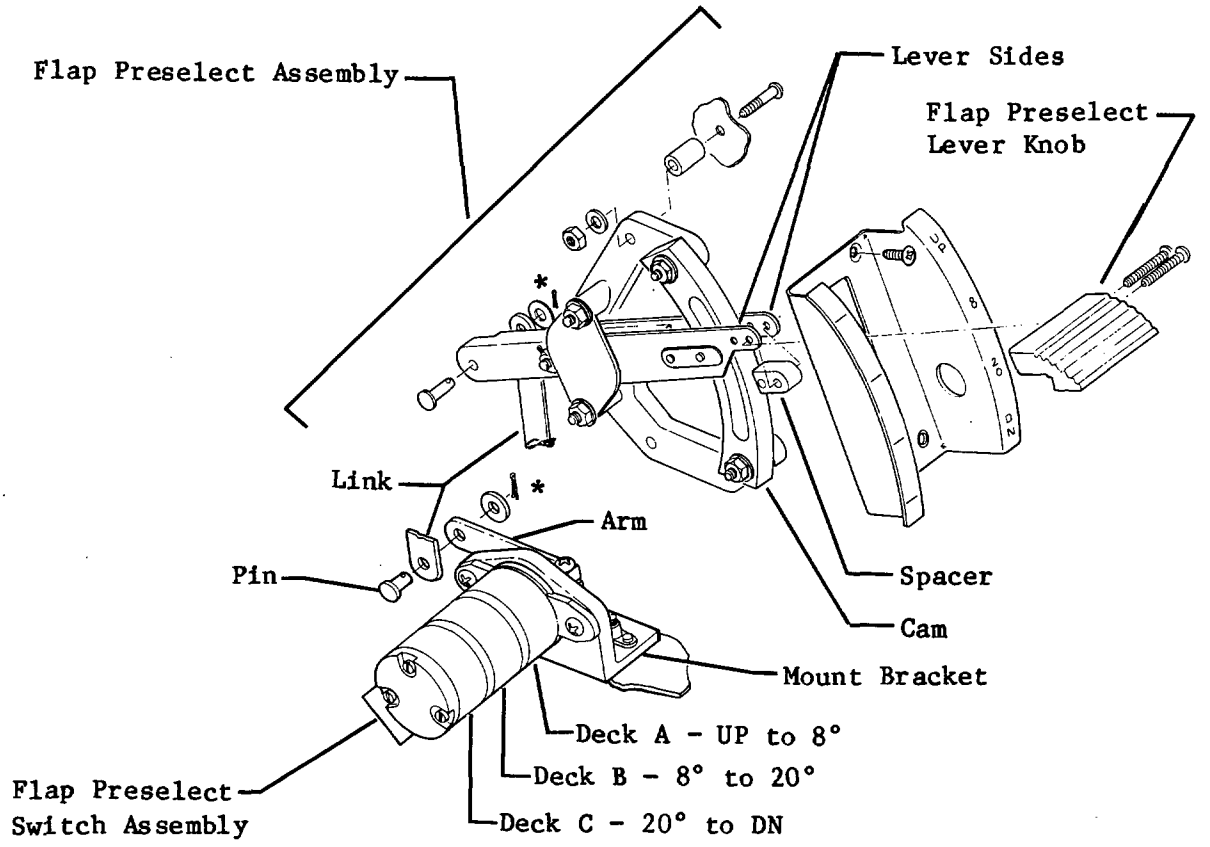
(5) Adjust Deck "B" switch as follows:
(a) Connect VOM between termianls NO and C.
(b) Set switch arm at 91° on protractor.
(c) Loosen lock screw on adjustment screw "A" and rotate adjustment screw cw or ccw until switch closes.
(d) Rotate switch arm toward 0° and check switch opens. This verifies that the internal wiper has been positioned correctly and is not 180° out-of-rig.
(e) Set switch arm at 180° on protractor.
(f) Loosen lock screw on adjustment screw "B" and rotate adjustment screw cw or ccw until switch opens.
(g) Rotate switch arm ccw to 0° on protractor.
(h) Rotate switch arm cw from 0° to 270° and ccw back to 0°.
(i) Check that switch closes at 91° ±2° and opens at 180° ±10°.
(j) The switch must be closed between 91° to 180° travel range.
(k) Tighten lock screws on adjustment screws "A" and "B".

(6) Adjust Deck "C" switch as follows:
(a) Connect VOM between terminals NO and C.
(b) Set switch arm at 107° on protractor.
(c) Loosen lock screw on adjustment screw "A" and rotate adjustment screw cw or ccw until switch closes.
(d) Rotate switch arm toward 0° and check switch opens. This verifies that the internal wiper has been positioned correctly and is not 180° out-of-rig.
(e) Set switch arm at 180° on protractor.
(f) Loosen lock screw on adjustment screw "B" and rotate adjustment screw cw or ccw until switch opens.
(g) Rotate switch arm ccw to 0° on protractor.
(h) Rotate switch arm cw from 0° to 270° and ccw back to 0°.
(i) Check that switch close at 107° ±2° and opens at 180° ±10°.
(j) The switch must be closed between 107° to 180° travel range.
(k) Tighten lock screws on adjustment screws "A" and "B".

(7) After initial adjustment is completed, procede with 2.B.(3) thru 2.B.(9).



*On Aircraft 35-549 thru 35-588, 35-601 and Subsequent, 36-054, 36-056 and Subsequent and prior Aircraft modified per AMK84-7, "Replacement of Flap Preselect Linkage," locate washers between arms and cotter pins.

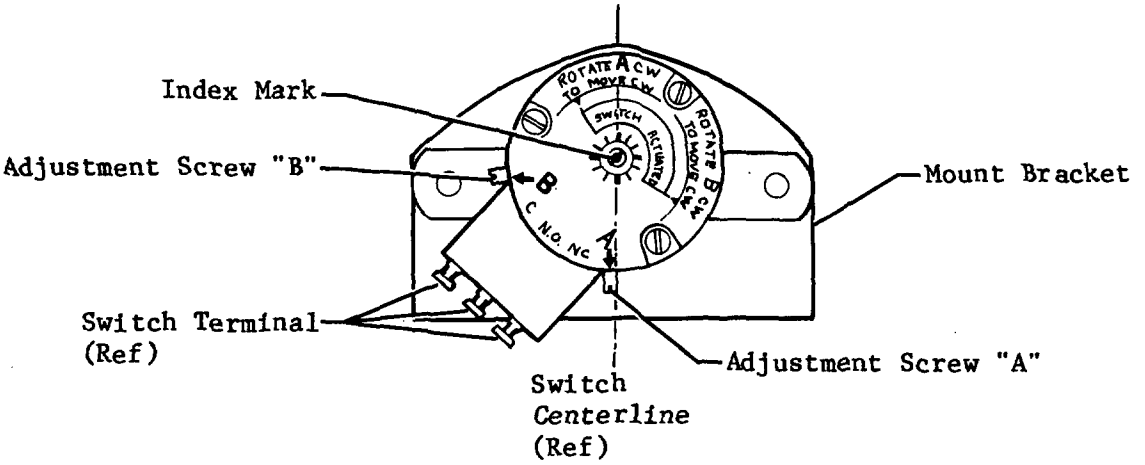


Detail A

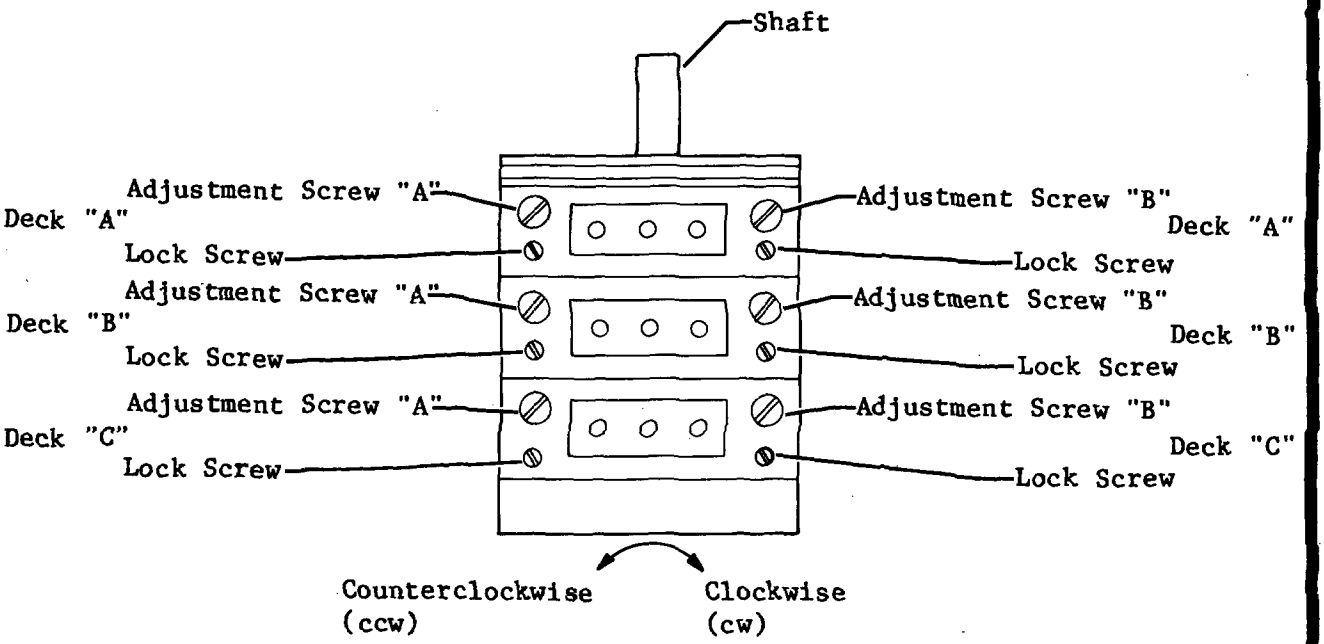
Preselect Assembly and Preselect Switch Installation
Figure 201

16-10B-2

End View of Switch Mounted on Bracket

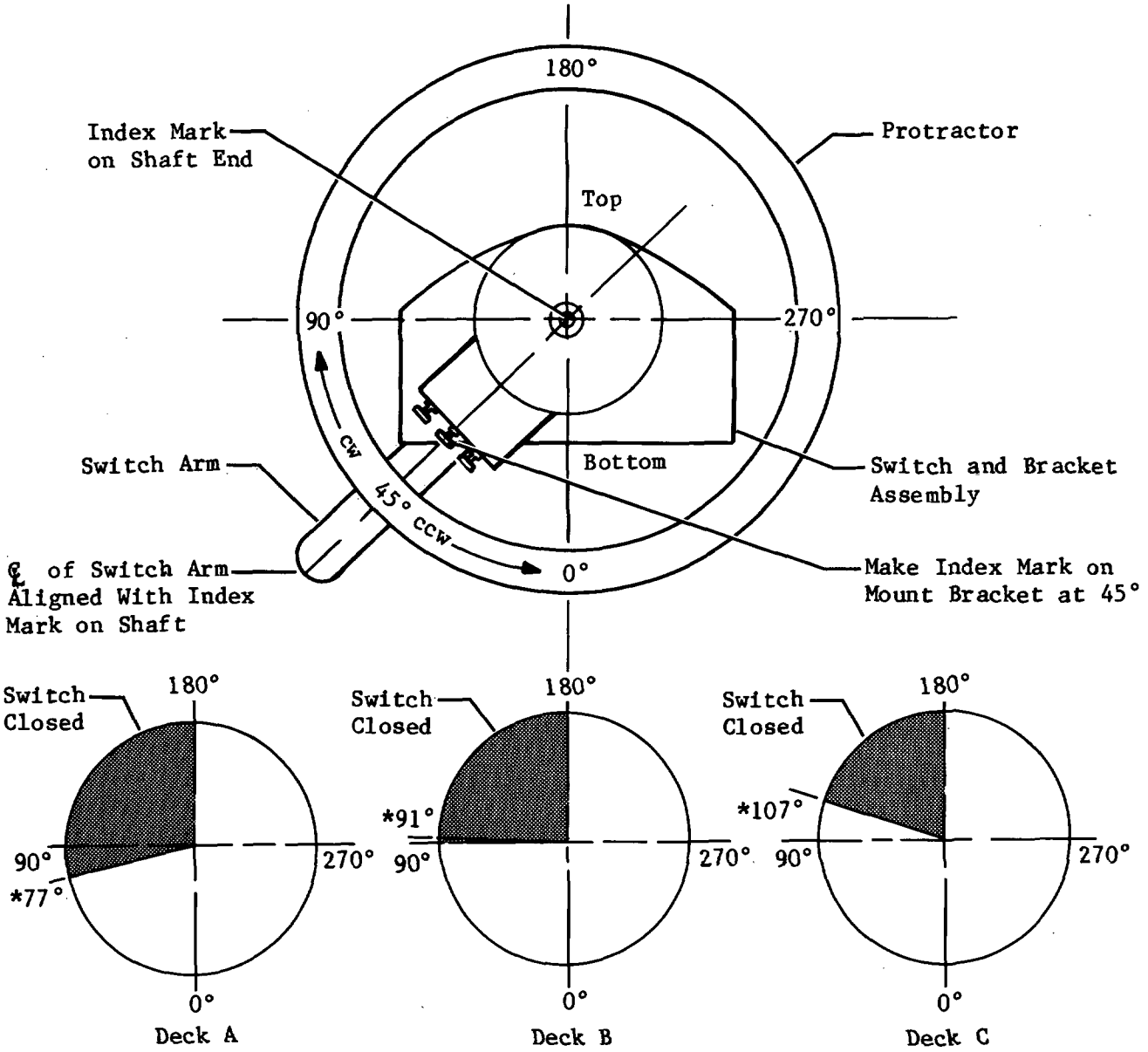


Top View of Switch Looking Down (Rotated to 180° on protractor)



Preselect Switch Adjustment
Figure 202 (Sheet 1 of 2)

LES-FT-1333
LES-FT-470



NOTE: ° All internal switch deck adjustments will fall within the shaded area.
* The 77°, 91°, 107°, and 180° positions are switch arm positions for initial adjustments on each deck respectively. The 71°, 91°, and 107° positions coincide with the 8°, 20°, and 40° flap preselect handle positions.

Preselect Switch Adjustment
Figure 202 (Sheet 2 of 2)

LES-FT-1333
LES-FT-470

GATED FLAP POSITION SWITCHES - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	To measure angles of control surfaces.
Digital Protractor (DP-45) (Alternate)	02475-01	Lucas & Schaebitz Pennsauken, NJ	To measure angles of control surfaces.
Multimeter	Model 260	Simpson Co. Elgin, IL	Measure resistance.

2. Removal/Installation

NOTE: Each flap position switch assembly consists of a switch, mounting bracket, and lever, preassembled and adjusted at the Learjet factory. Only minor adjustments may be required after installation of the switch assembly.

A. Remove Flap Position Switch Assembly (S157, LH; S156, RH) (See Figure 201.)

WARNING: BLOCK SPOILERS UP TO PREVENT INADVERTENT SPOILER RETRACTION AND BODILY INJURY.

(1) With hydraulic and electrical power supplied to aircraft, fully extend spoilers and flaps.
(2) Depressurize hydraulic system. (Refer to Chapter 12.)
(3) Remove electrical power from aircraft.
(4) Disconnect and identify electrical wiring at position switch assembly.
(5) Remove attaching parts securing position switch lever to position switch link.
(6) Remove attaching parts securing position switch assembly mounting bracket to aircraft structure and remove position switch assembly from aircraft.

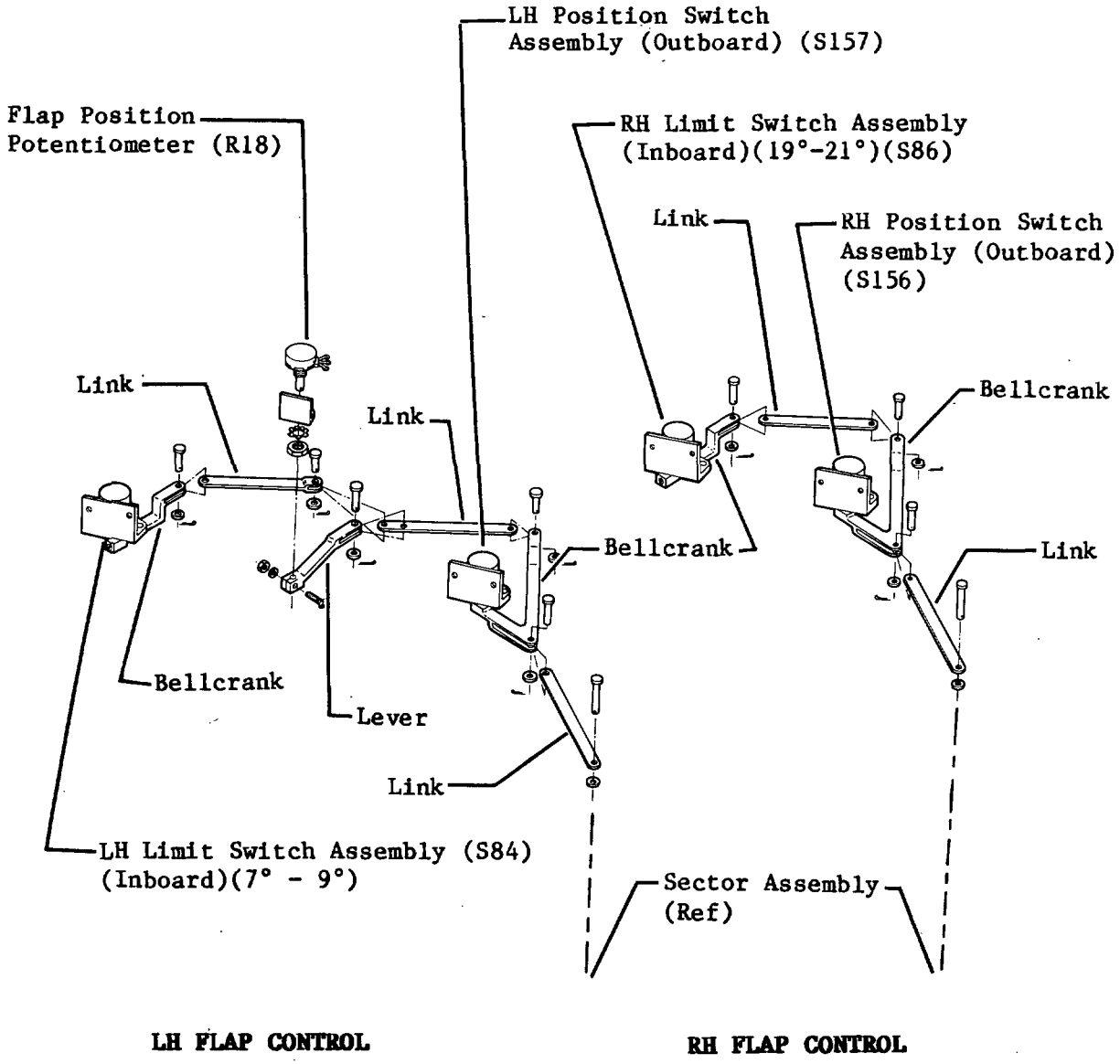
B. Install Flap Position Switch Assembly (S157, LH; S156, RH) (See Figure 201.)
(1) Place position switch assembly on aircraft structure and secure with attaching parts.
(2) Secure position switch lever to position switch link with attaching parts.
(3) Identify and connect electrical wiring at position switch assembly.
(4) Adjust position switch assembly. (Refer to Adjustment/Test, this section.)
(5) Perform functional test of flap system. (Refer to 27-51-00, Adjustment/Test.)
(6) Pressurize hydraulic system.
(7) Restore electrical power to aircraft.
(8) Stow flaps and spoilers in full retract position.
(9) Restore aircraft to normal.

3. Adjustment/Test

A. Adjustment of Flap Position Switch (S157, LH; S156, RH)

WARNING: BLOCK SPOILERS UP TO PREVENT INADVERTENT SPOILER RETRACTION AND BODILY INJURY.

(1) With hydraulic and electrical power supplied to aircraft, fully extend spoilers and flaps.



Flap Position Switch Installation
Figure 201

(2) Disconnect both flap actuators from flap sectors.
(3) Manually raise flaps and block in place.
(4) Attach clinometer to flaps.
(5) Remove nose compartment access doors to gain access to stall warning computer (E31). Disconnect electrical connectors P914 and P913 from stall warning computer (E31).
(6) Hold RH flap in 0° position, probe between connector P914 pin U and ground with ohmmeter, and observe an open circuit. Lower RH flap to 3 (±0.5)° and observe continuity indication on ohmmeter.
(7) Repeat step (6) for LH position switch; use LH flap and probe between connector P913 pin U and ground.

NOTE:
- If indications in steps (6) and (7) are correct, proceed to step (9).
- The flap position switches are preset internally and bellcranks positioned correctly at the factory; however, after installation it may be necessary to make a minor adjustment in accordance with step (8).

(8) If indications in steps (6) and (7) are not correct, proceed as follows:
(a) Gain access to switch that is out of tolerance.
(b) Adjust inboard adjusting screw of lower deck until switch is within tolerance.
(c) Repeat steps (6) and (7) to ensure that both switches are within tolerance.
(9) Position RH flap in 0° position, probe between connector P914 pin E and ground, and observe an open circuit. Lower flaps to 13 (±0.5)° and observe continuity.
(10) Repeat step (9) for LH position switch; use LH flap and probe between connector P913 pin E and ground.

NOTE: If indications in steps (9) and (10) are correct, proceed with step (12).

(11) If indications in steps (9) and/or (10) are not correct, proceed as follows:
(a) Gain access to switch that is out of tolerance.
(b) Adjust inboard screw of middle deck until switch is within tolerance.
(c) Repeat steps (6) thru (10) to ensure that all decks on both switches are within tolerance.
(12) Position RH flap in 0° position, probe between connector P914 pin F and ground with ohmmeter, and observe an open circuit. Lower RH flap to 25 (±0.5)° and observe continuity.
(13) Repeat step (12) for LH position switch; use LH flap and probe between connector P913 pin F and ground.

NOTE: If indications in steps (12) and (13) are correct, proceed with step (15).

(14) If indications in steps (12) and/or (13) are not correct, proceed as follows:
(a) Gain access to switch that is out of tolerance.
(b) Adjust inboard adjusting screw of upper deck until switch is within tolerance.
(c) Repeat steps (6) thru (13) to ensure that all decks on both switches are within tolerance.
(15) If any one tolerance cannot be maintained on a switch, that switch must be replaced.
(16) Connect connectors P914 and P913 to stall warning computer (E31) and install nose compartment access doors.
(17) Lower flaps. Install flap actuators to flap sectors.
(18) Perform flap system functional test. (Refer to 27-51-00.)
(19) Restore aircraft to normal.

FLAP LIMIT SWITCHES - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	To measure angles of control surfaces.
Digital Protractor (DP-45) (Alternate)	02475-01	Lucas & Schaebitz Pennsauken, NJ	To measure angles of control surfaces.
Multimeter	Model 260	Simpson Co. Elgin, IL	Measure resistance.
Potting Compound	MIL-S-8516	Commercially Available	Protect electrical components.

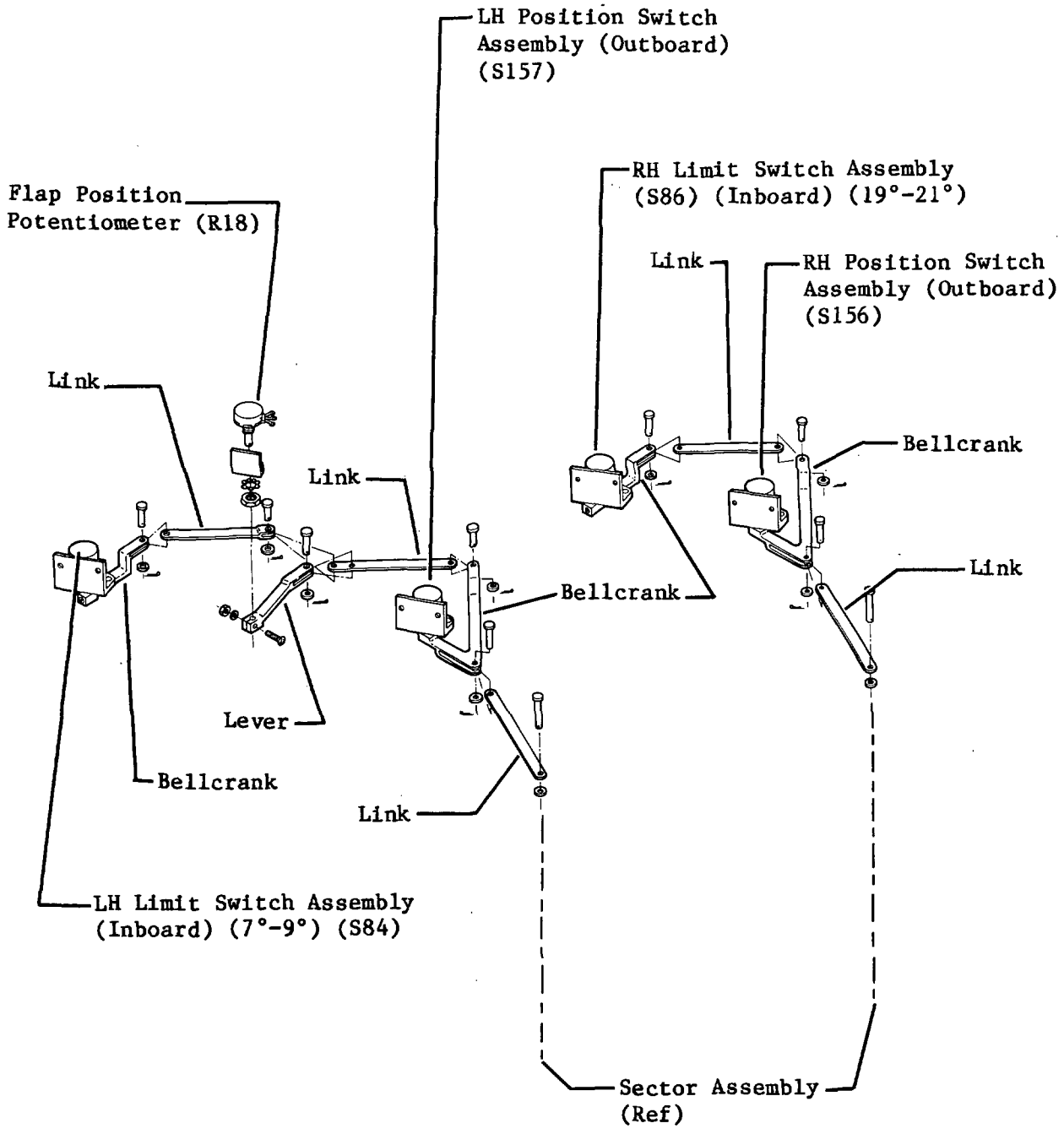
2. Removal/Installation

NOTE: Each switch assembly consists of a switch, mounting bracket, and lever, preassembled and adjusted at the Learjet factory. Only minor adjustments may be required after installation of the switch assembly.

A. Remove Flap Limit Switch Assembly (See Figure 201.)

WARNING: LOCK SPOILERS UP TO PREVENT INADVERTENT SPOILER RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps to gain access to flap limit switch assemblies at WS 61 (LH) and WS 66 (RH).
(2) Remove cotter pin, washers, and pin from point where flap limit switch lever attaches to position switch bellcrank link (RH) or to position potentiometer lever link (LH).
(3) Remove screws attaching mounting bracket on wing spar 8.
(4) Remove potting compound from switch terminals and disconnect electrical wires from flap limit switch.



Flap Limit Switches Installation
Figure 201

B. Install Flap Limit Switch Assembly (See figure 201.)

WARNING: BLOCK SPOILERS UP TO PREVENT INADVERTENT SPOILER RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps to gain access to flap limit switch area at WS 61 (LH) and WS 66 (RH).
(2) Ensure that switch terminals are clean and free of oil, dirt, grease, and potting compound.

NOTE: Deck A of flap limit switches is the lower deck, deck B is the middle deck, and deck C is the upper deck (not used).

(3) Connect electrical wires to flap limit switch. (Refer to 35/36 Wiring Manual.)
(4) Apply MIL-S-8516 potting compound over switch terminals and wires. Do not allow gaps and voids to form when applying potting compound. Allow potting compound to cure in accordance with manufacturer's instructions.
(5) Position flap limit switch mounting bracket on spar 8 and secure with attaching parts.
(6) Install pin, washers, and cotter pin at point where lever attaches to position switch bellcrank link (RH) or to position potentiometer lever link (LH).

3. ADJUSTMENT/TEST

A. Rig Flap Limit Switches

WARNING: BLOCK SPOILERS UP TO PREVENT INADVERTENT SPOILER RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps. Block spoilers up.
(2) Disconnect flap actuator from flap sectors.
(3) Manually raise flaps and block in place.
(4) Attach clinometer to flaps.
(5) Locate connector P424 in RH wheel well and disconnect. Locate connector P191 at flap control valve and disconnect.
(6) Using an ohmmeter, probe between connector P424 pin 22 and P191 pin A. Observe continuity.
(7) Unblock flaps and move flaps down until ohmmeter indicates open circuit. The clinometer shall read between 7° and 7-1/2° flaps down.
(8) If the indication in step (7) is not correct, position flaps between 7° and 7-1/2° flaps down, gain access to LH limit switch, and adjust the inboard adjusting screw on the lower deck until ohmmeter indicates open circuit. Repeat step (7) after each adjustment.
(9) Using an ohmmeter, probe between connector P191 pin B and P424 pin 22. Observe open circuit.
(10) Move flaps down until ohmmeter indicates continuity. The clinometer shall read between 8-1/2° and 9° flaps down.
(11) If the indication in step (10) is not correct, position flaps between 8-1/2° and 9° flaps down, gain access to LH limit switch, and adjust the outboard adjusting screw on the middle deck until ohmmeter indicates continuity. Repeat step (11) after each adjustment.
(12) When the LH limit switch is adjusted correctly in step (11), repeat steps (6) thru (8) to ensure that the lower deck was not disturbed.
(13) Using an ohmmeter, probe between connector P424 pin 23 and P191 pin A. Observe continuity.
(14) Move flaps down until ohmmeter indicates open circuit. The clinometer shall read between 19° and 19-1/2° flaps down.

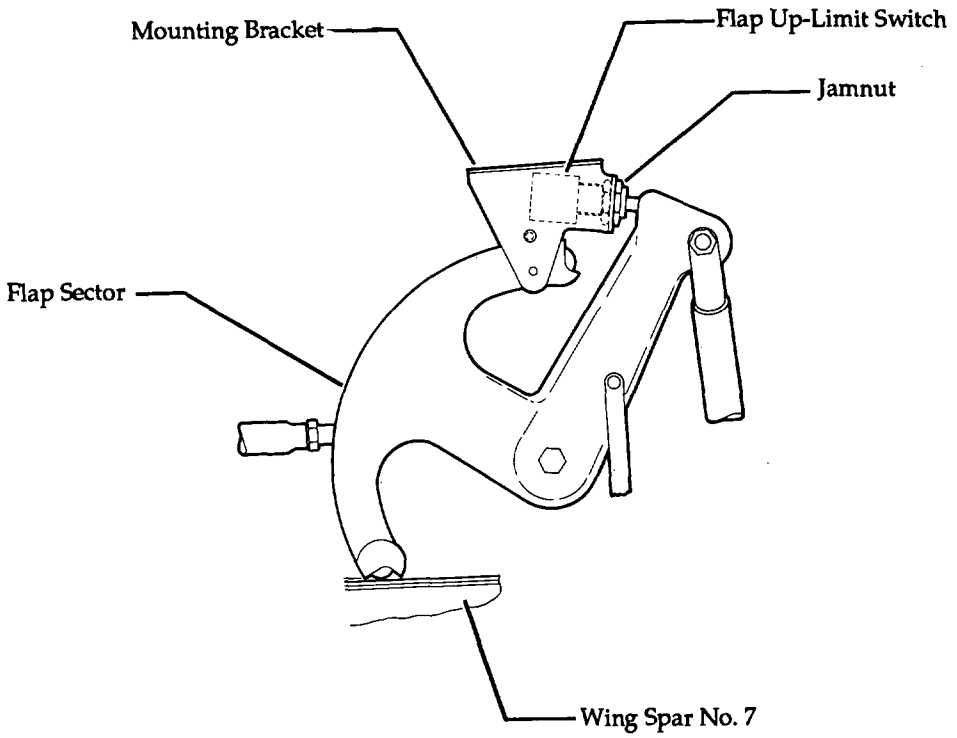
(15) If the indication in step (14) is not correct, position flaps between 19° and 19-1/2° flaps down, gain access to RH limit switch, and adjust the inboard adjusting screw on the lower deck until ohmmeter indicates open circuit. Repeat step (14) after each adjustment.
(16) Using an ohmmeter, probe between connector P191 pin B (flap control valve) and P424 pin 23. Observe open circuit.
(17) Move flaps down until ohmmeter indicates continuity. The clinometer shall read between 20-1/2° and 21° flaps down.
(18) If the indication in step (17) is not correct, position flaps between 20-1/2° and 21°, gain access to RH limit switch, and adjust the outboard adjusting screw on the middle deck until ohmmeter indicates continuity. Repeat step (17) after each adjustment.
(19) When the RH limit switch is adjusted correctly, in step (18), repeat steps (13) thru (15) to ensure that the lower deck was not disturbed.
(20) Connect connector P191 to flap control valve.
(21) Connect connector P424 to J424.
(22) Connect flap actuators to flap sectors.
(23) Remove blocks from spoilers and retract spoilers.
(24) Perform flap system functional test. (Refer to 27-51-00.)

FLAP UP-LIMIT SWITCH - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

A. Remove Flap Up-Limit Switch (See figure 201.)
   (1) Remove power from aircraft.
   (2) Remove flap sector access cover from left wing.
   (3) Remove attaching parts and remove flap up-limit switch from mounting bracket.
   (4) Disconnect electrical wiring from flap up-limit switch.
   (5) Remove switch from aircraft.

B. Install Flap Up-Limit Switch (See figure 201.)
   (1) Connect electrical wiring to flap up-limit switch.
   (2) Position flap up-limit switch on mounting bracket and secure with attaching parts.
   (3) Restore electrical power to aircraft.
   (4) Perform Gated Flap Control System Functional Test. (Refer to 27-51-00.)



Flap Up-Limit Switch Installation
Figure 201

EFFECTIVITY: 35-417, 35-419, 35-477, 35-479, 35-483 THRU 35-531, AND 36-051 THRU 36-053, NOT MODIFIED PER SB 35/36-27-15, "INSPECT FLAP SECTOR MOUNTING BRACKETS FOR CRACKS AND FLAP SYSTEM FOR INTERFERENCE.

FLAP CONTROL VALVE - MAINTENANCE PRACTICES

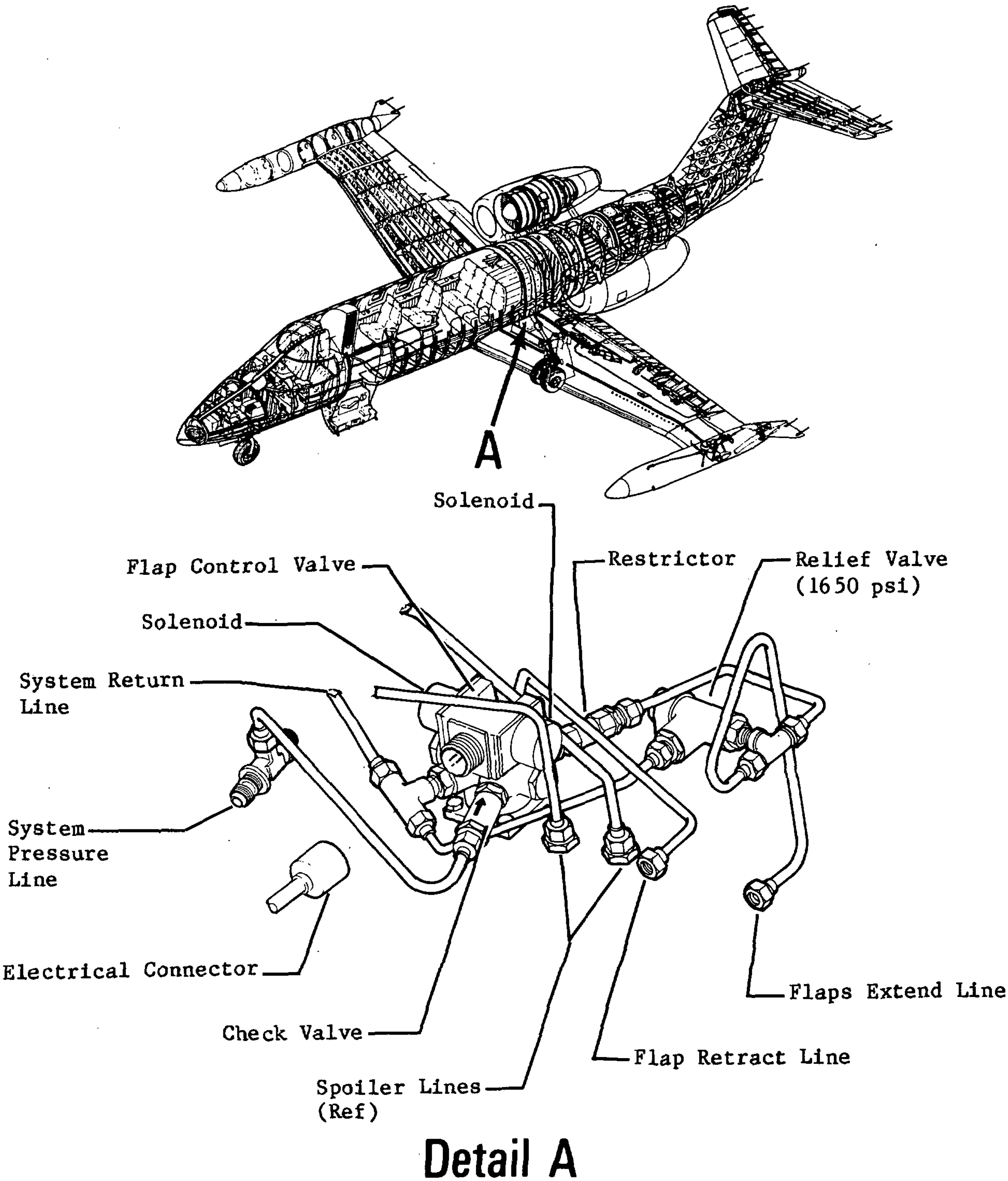
1. REMOVAL/INSTALLATION

A. Remove Flap Control Valve (See figure 201.)

(1) Depressurize hydraulic system. (Refer to Chapter 29.)
(2) Remove electrical power from aircraft.
(3) Lower LH inboard main landing gear door. (Refer to Chapter 32.)
(4) Disconnect electrical connector from flap control valve.
(5) Disconnect hydraulic pressure line from check valve at system pressure port of control valve and plug hydraulic line.
(6) Disconnect and plug lines from tee at system return port of control valve.
(7) Disconnect and plug line at flap up port of control valve.
(8) Disconnect and plug hydraulic line from restrictor on flap down port of control valve.
(9) Remove attaching parts and flap control valve from mounting bracket. Remove flap control valve from aircraft.
(10) Remove elbow, tee, restrictor, and union from ports of flap control valve. Retain parts for installation on new valve.

B. Install Flap Control Valve (See figure 201.)

(1) Clean and inspect check valve, union, tee, and restrictor removed from old flap control valve. If serviceable, install in new flap control valve (use new O-rings and packings).
(2) Position flap control valve on mounting bracket. Ensure that hydraulic fittings are correctly positioned to facilitate proper installation.
(3) Secure flap control valve to mounting bracket with attaching parts.
(4) Remove plugs from hydraulic lines and connect hydraulic lines to respective ports of flap control valve.
(5) Connect electrical connector to flap control valve and safety wire.
(6) Restore electrical power to aircraft.
(7) Bleed hydraulic lines, pressurize hydraulic system, check for leaks, and check flap operation.
(8) Connect mechanical linkage to LH inboard main landing gear door.



11-19C-1

Flap Control Valve Installation
Figure 201

FLAP ACTUATOR - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

NOTE: The following procedure is applicable to both RH and LH flap actuators.

A. Remove Flap Actuator (See figure 201.)
(1) Depressurize hydraulic system. (Refer to Chapter 29.)
(2) Remove flap actuator access cover.
(3) Disconnect and cap hydraulic lines at actuator.
(4) Remove attaching parts from attach bracket and flap sector and remove flap actuator.

B. Install Flap Actuator (See figure 201.)
(1) Position flap actuator between attach bracket and flap sector and secure with attaching parts.
(2) Remove caps and connect hydraulic lines to actuator.
(3) Rig flap system. (Refer to 27-51-00.)
(4) Bleed and pressurize hydraulic system. (Refer to Chapter 29.)
(5) Install access cover.

2. APPROVED REPAIRS

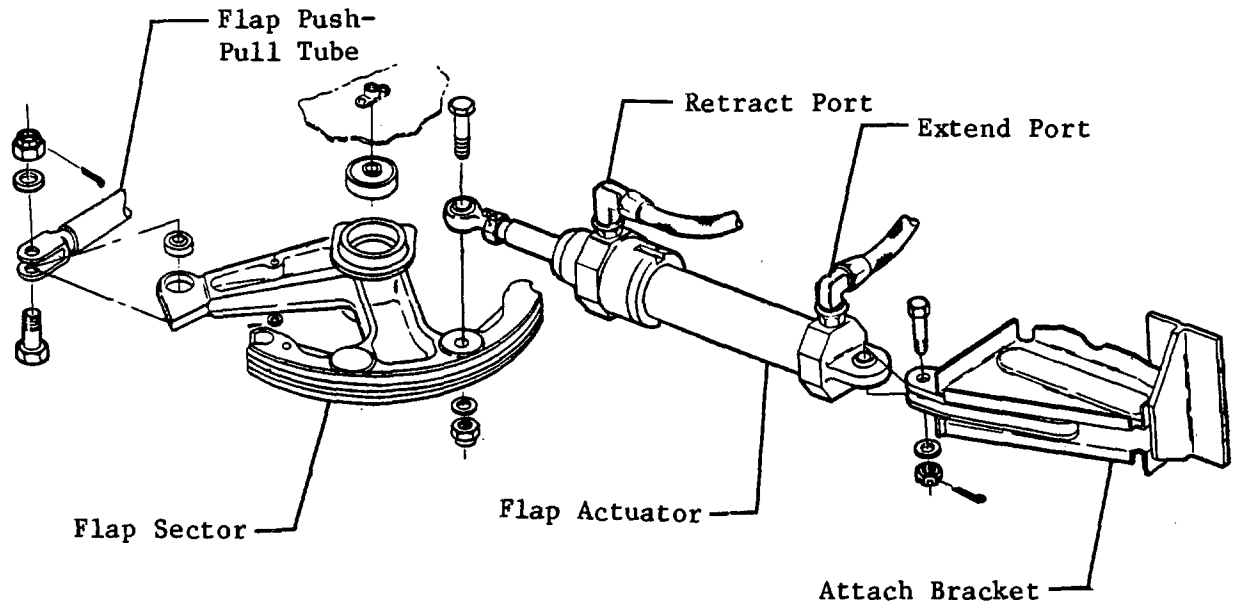
NOTE: ° The following procedure lists disassembly and assembly of flap actuator necessary to replace O-rings, backup rings, and felt wiper. New O-rings, backup rings, and felt wiper must be installed at each assembly. These new items are to be immersed in MIL-H-5606 hydraulic fluid immediately prior to installation.

° All O-ring grooves are to be free of gouges, scratches, and burrs. Assure all items are thoroughly clean prior to installation.

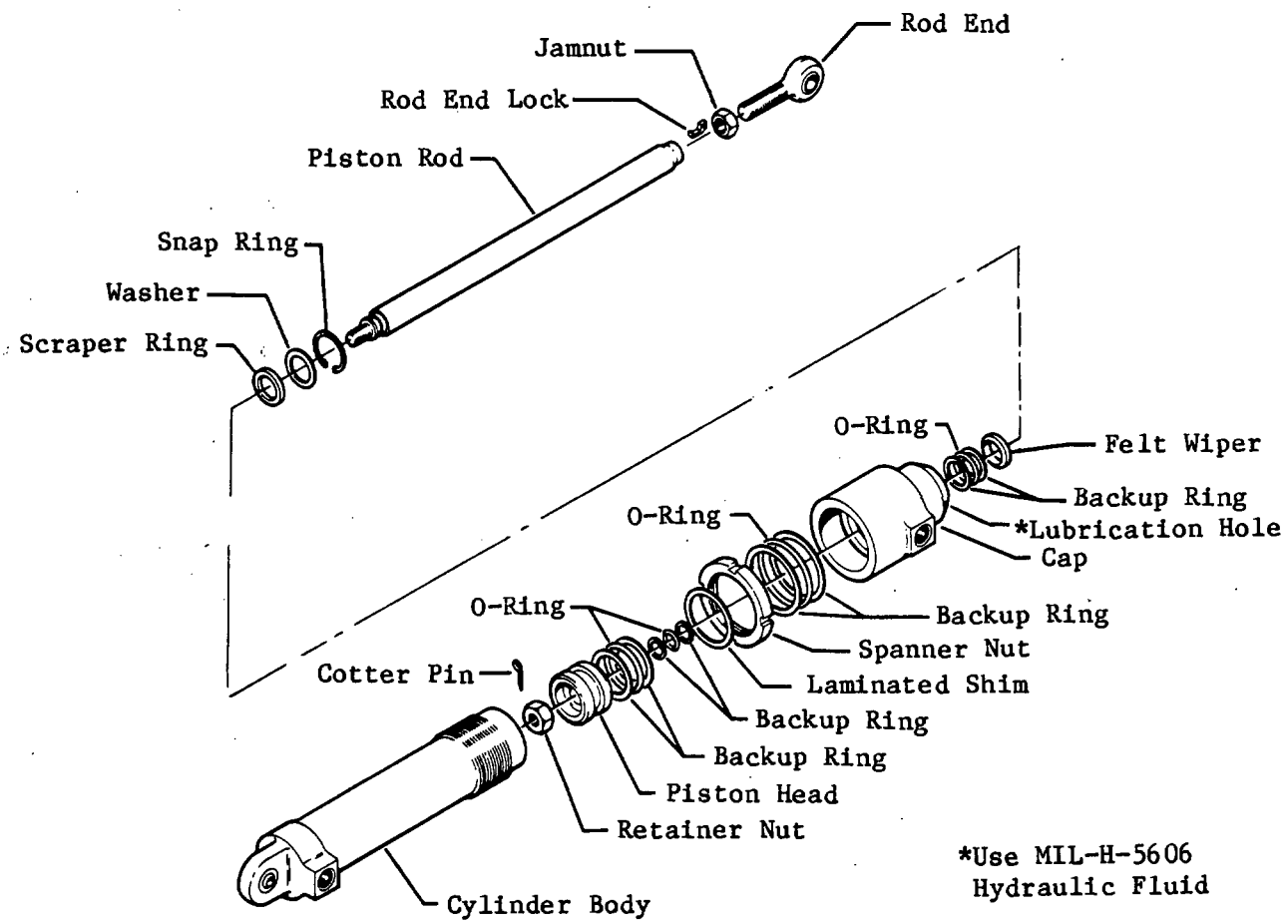
A. Disassemble Flap Actuator (See figure 202.)
(1) Cut safety wire and loosen jamnut sufficiently to release rod end lock. Remove rod end from piston rod.
(2) Cut safety wire and loosen spanner nut. Remove piston rod from cylinder body.
(3) Remove cotter pin and retainer nut securing piston head to piston rod.
(4) Remove snap ring from cap. Remove all backup rings, O-rings, washer, laminated shim, scraper ring, and felt wiper from piston rod and piston head, cylinder body, and cap.
(5) Clean all parts in Stoddard solvent or equivalent.
(6) Inspect cylinder body, cap, piston rod, and piston head for nicks, scratches, or excessive wear. Buff out minor defects with an oil stone, then polish with crocus cloth.

B. Assemble Flap Actuator (See figure 202.)
(1) Immerse all internal parts in MIL-H-5606.
(2) Assemble O-ring and backup ring on piston rod.

(3) Install piston head on piston rod and secure with retainer nut and cotter pin. Assemble O-ring and backup rings on piston head.
(4) Apply a thin coat of MIL-H-5606 hydraulic fluid to piston head and insert piston rod with piston head attached into cylinder body.
(5) Assemble O-rings, backup rings, laminated shim, and felt wiper in cap. Screw cap on cylinder body as far as possible, assuring that extend and retract ports are in line.
(6) Lubrication of the felt wiper is required every 600 hours. Use only MIL-H-5606 hydraulic fluid.
(7) Tighten spanner nut against cap and safety wire.
(8) Assemble scraper ring, washer, and snap ring in cap. Install rod end with jamnut and rod end lock in piston rod.
(9) Connect actuator to a hydraulic test bench. Pressure check actuator, both extend and retract positions, to 2250 psi for 3 minutes. Leakage should not exceed one drop in 3-minute period.
(10) With actuator retracted, adjust for 12.950 (±0.010) inches between centerline of actuator bearing and rod end. Safety wire rod end lock.



Flap Actuator Installation
Figure 201



Flap Actuator Assembly
Figure 202

FLAP POSITION POTENTIOMETER - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

A. Remove Flap Position Potentiometer (See figure 201.)

WARNING: BLOCK SPOILER UP TO PREVENT INADVERTENT SPOILER RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps to gain access to flap position potentiometer on wing spar 8, at left WS 66.
(2) Tag and disconnect electrical wires from flap position potentiometer.
(3) Loosen set screw that secures lever to flap position potentiometer shaft and remove lever from shaft.
(4) Loosen jamnut securing potentiometer in bracket and remove flap position potentiometer.

B. Install Flap Position Potentiometer (See figure 201.)

WARNING: BLOCK SPOILER UP TO PREVENT INADVERTENT SPOILER RETRACTION AND BODILY INJURY.

(1) Extend spoilers and lower flaps to gain access to flap position potentiometer installation area.
(2) Install potentiometer in bracket and secure with jamnut.
(3) Install lever on potentiometer shaft and secure with set screw. Check that lever, link, and bellcrank are not misaligned or binding.
(4) Remove tags and connect electrical wires to flap position potentiometer.

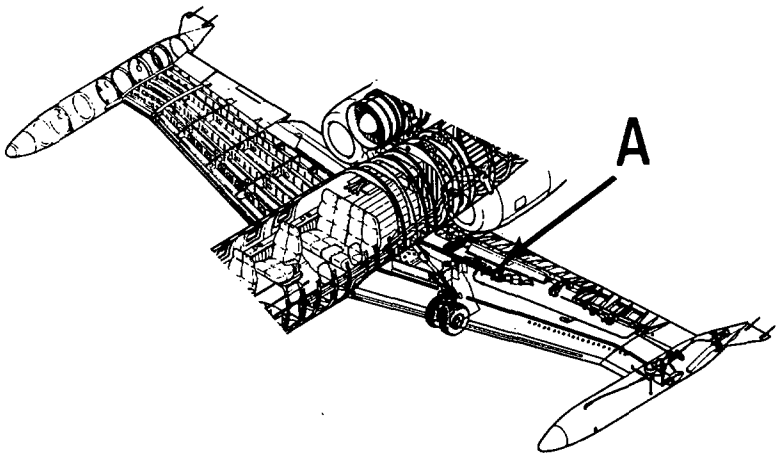
2. ADJUSTMENT/TEST

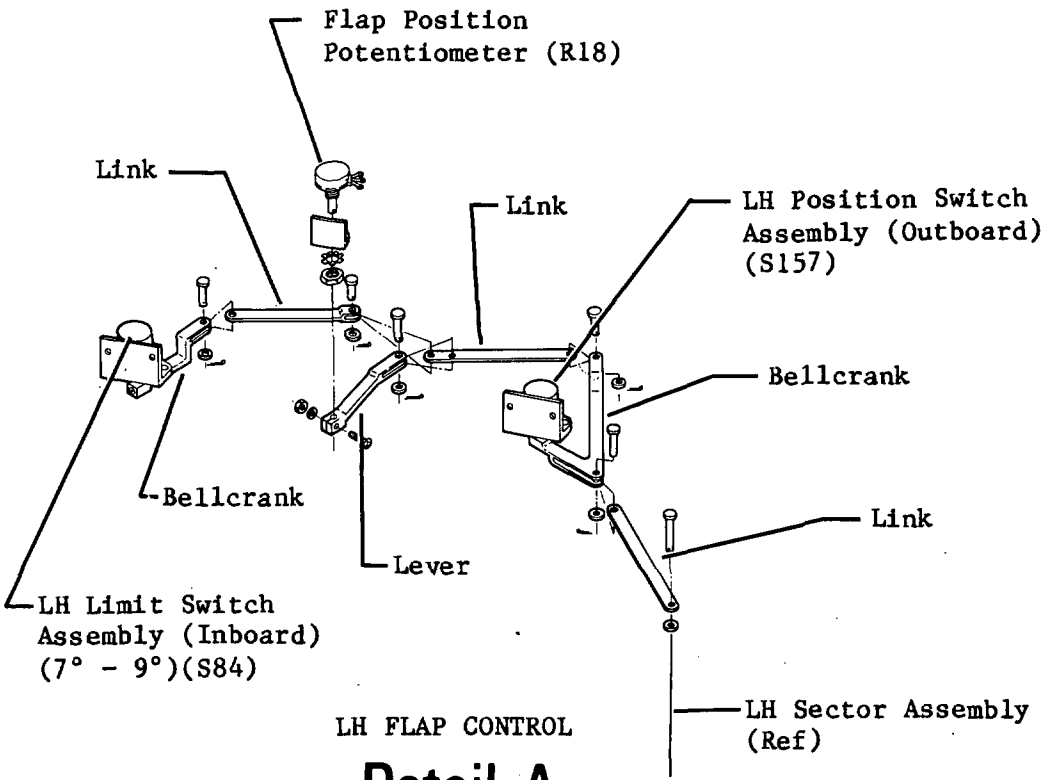
NOTE: Flap position potentiometer rigging will be required only when potentiometer has been disconnected from lever. Rigging is accomplished by rotating potentiometer shaft.

A. Rig Flap Position Potentiometer

NOTE: The flaps must be properly rigged prior to adjusting flap position potentiometer.

(1) Attach a clinometer flap and lower flaps to 20° (±0.5°).
(2) Loosen screw and nut securing lever to potentiometer shaft.
(3) Rotate shaft, using a screwdriver in slot at bottom of potentiometer shaft, until flap position indicator in instrument panel indicates 20°.
(4) Secure lever screw and nut, assuring that vertical location of lever on shaft does not induce binding between link and lever.





LH FLAP CONTROL

Detail A

Flap Position Potentiometer Installation
Figure 201

8-6C-3

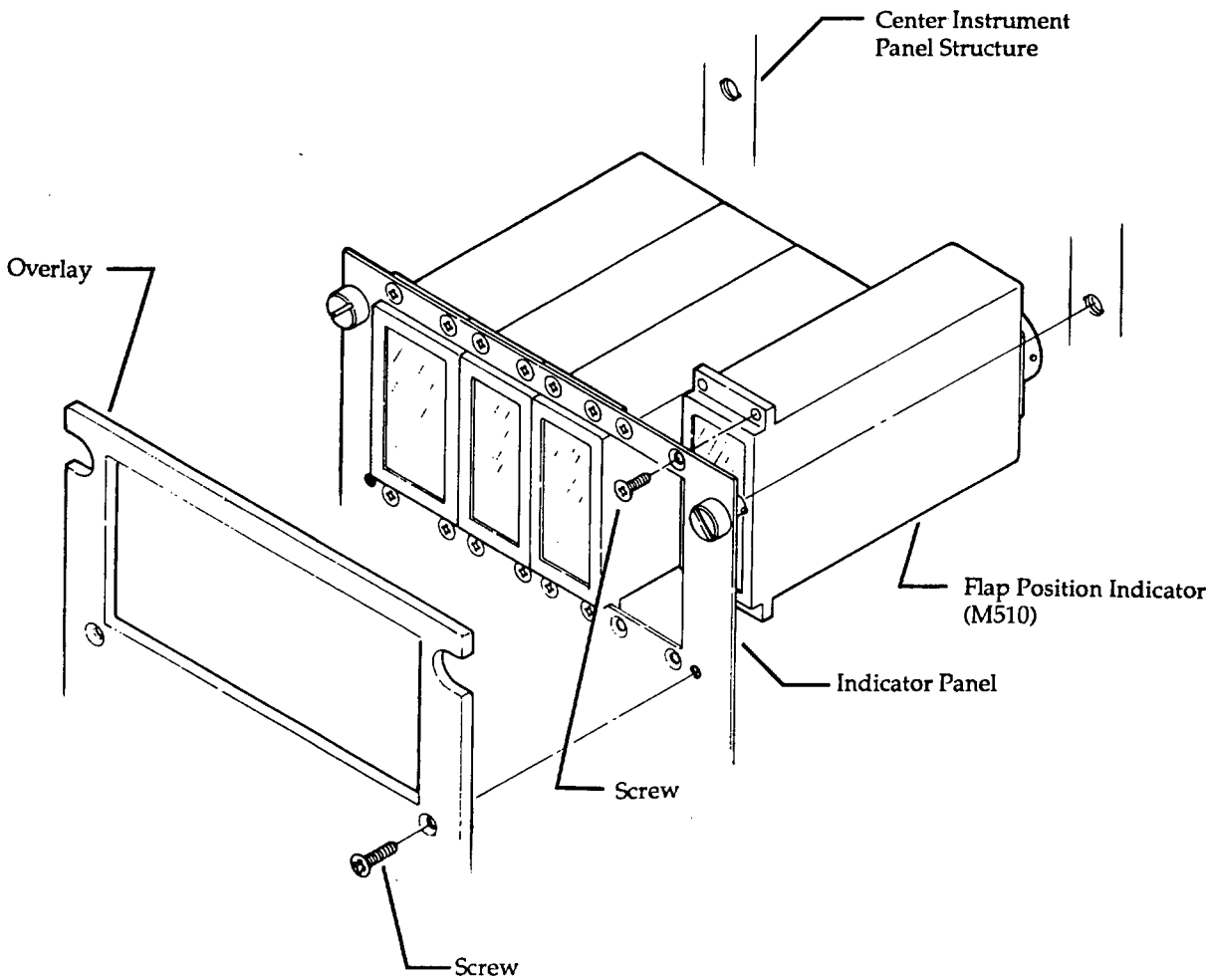
GATED FLAP POSITION INDICATOR - MAINTENANCE PRACTICES

1. **Removal/Installation**
   A. Remove Flap Position Indicator (See figure 201.)
      (1) Remove electrical power from aircraft.
      (2) Loosen and remove attaching parts, and indicator from instrument panel.
      (3) Disconnect electrical connector from flap position indicator.
      (4) Remove attaching parts and remove flap position indicator.
   B. Install Flap Position Indicator (See figure 201.)
      (1) Position flap position indicator in indicator panel and secure with attaching parts.
      (2) Connect electrical connector to flap position indicator.
      (3) Position overlay, and indicator panel on instrument panel. Secure with attaching parts.
      (4) Restore electrical power to aircraft.
      (5) Perform gated flap system functional test. (Refer to 27-51-00.)

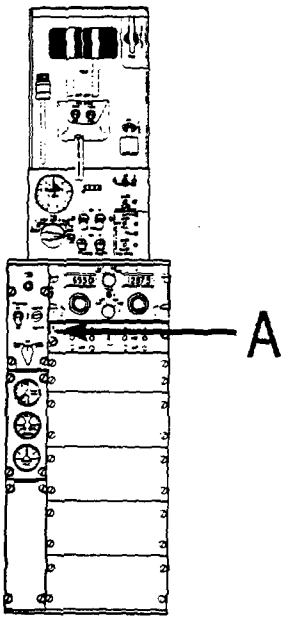
2. **Adjustment/Test** (See figure 202.)
   A. Calibrate flap position indicator as follows:

      NOTE:
      - The flap position indicator, located on the center instrument panel, indicates flap position. Input to the indicator is from a position transmitter attached to the LH flap sector.
      - Calibration of the flap position indicator is performed at the position indicator calibration assembly located inside the trim switch panel.

      (1) Rig flaps in accordance with procedures outlined in Flap System Adjustment/Test. (Refer to 27-51-00.)
      (2) Remove equipment as necessary from the pedestal adjacent to the trim switch panel. (On *Aircraft 35-417, 35-419, 35-477, 35-479, 35-483 thru 35-505, 36-051 thru 36-053, and aircraft not modified per AAK 83-2, "Installation of FC-530 Autopilot"*, refer to Chapter 31. On *Aircraft 35-506 and Subsequent, 36-054 and Subsequent and prior aircraft modified per AAK 83-2 "Installation of FC-530 Autopilot"*, refer to Chapter 39. This allows access to the position indicator calibration assembly.
      (3) Actuate flaps to full up position.
      (4) Adjust Zero Flap potentiometer to obtain flaps UP indication on the flap position indicator.
      (5) Actuate flaps to full down position.
      (6) Adjust Flap Down potentiometer to obtain flaps DN indication on flap position indicator.
      (7) Cycle flaps while checking respective indication on flap position indicator.
      (8) Readjust Flap Down potentiometer as necessary to obtain coordinated indications.
      (9) Replace and secure equipment in the pedestal.
      (10) Perform gated flap system functional test. (Refer to 27-51-00.)



Flap Position Indicator Installation
Figure 201



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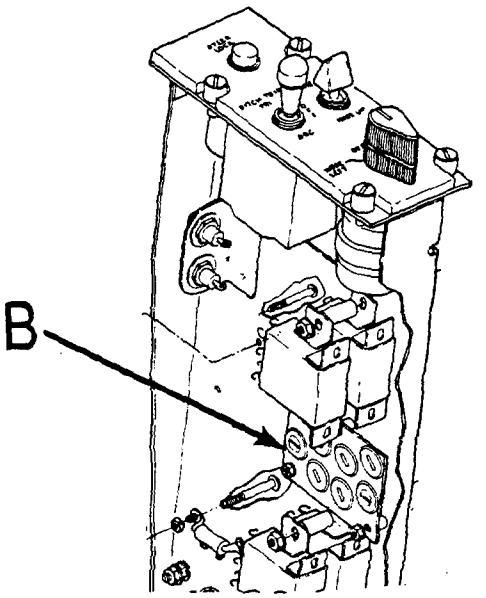
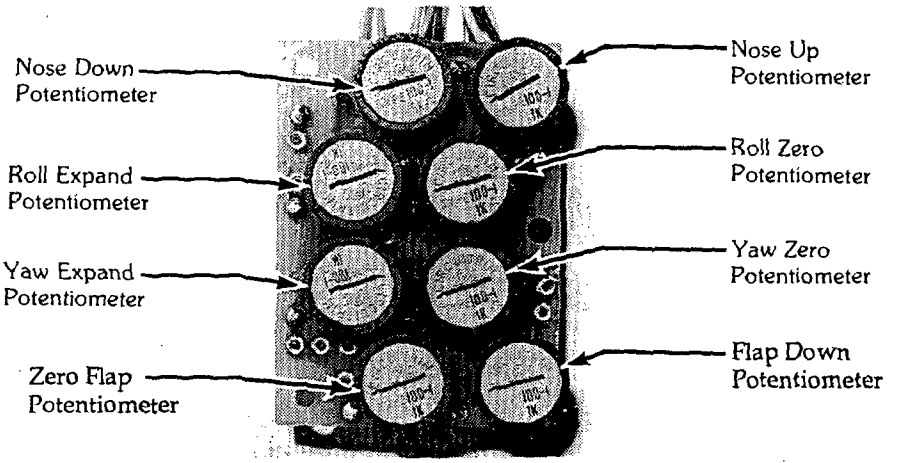


Detail A



ORIGINAL
As Received By
ATP

Detail B

Position Indicator Calibration Assembly
Figure 202

FLAPS SECTOR - MAINTENANCE PRACTICES

1. Removal/Installation

A. Removal of Flap Sector (See Figure 201.)

(1) Connect Electrical power source to aircraft.
(2) Connect external hydraulic power source to aircraft.
(3) Extend flaps and spoilers.
(4) Remove flap sector access cover from LH and RH wings.
(5) Disconnect flap actuator from flap sector. (Refer to 27-51-07, Removal /Installation.)
(6) Disconnect flap push-pull tube from flap sector. (Refer to 27-51-01, Removal /Installation.)
(7) Remove safety wire from flap sector bolt, remove bolt and flap sector from wing.

NOTE: It is not necessary to loosen flap cable turnbuckles to remove flap sectors. Turn flap sectors over and remove cables from flap sectors.

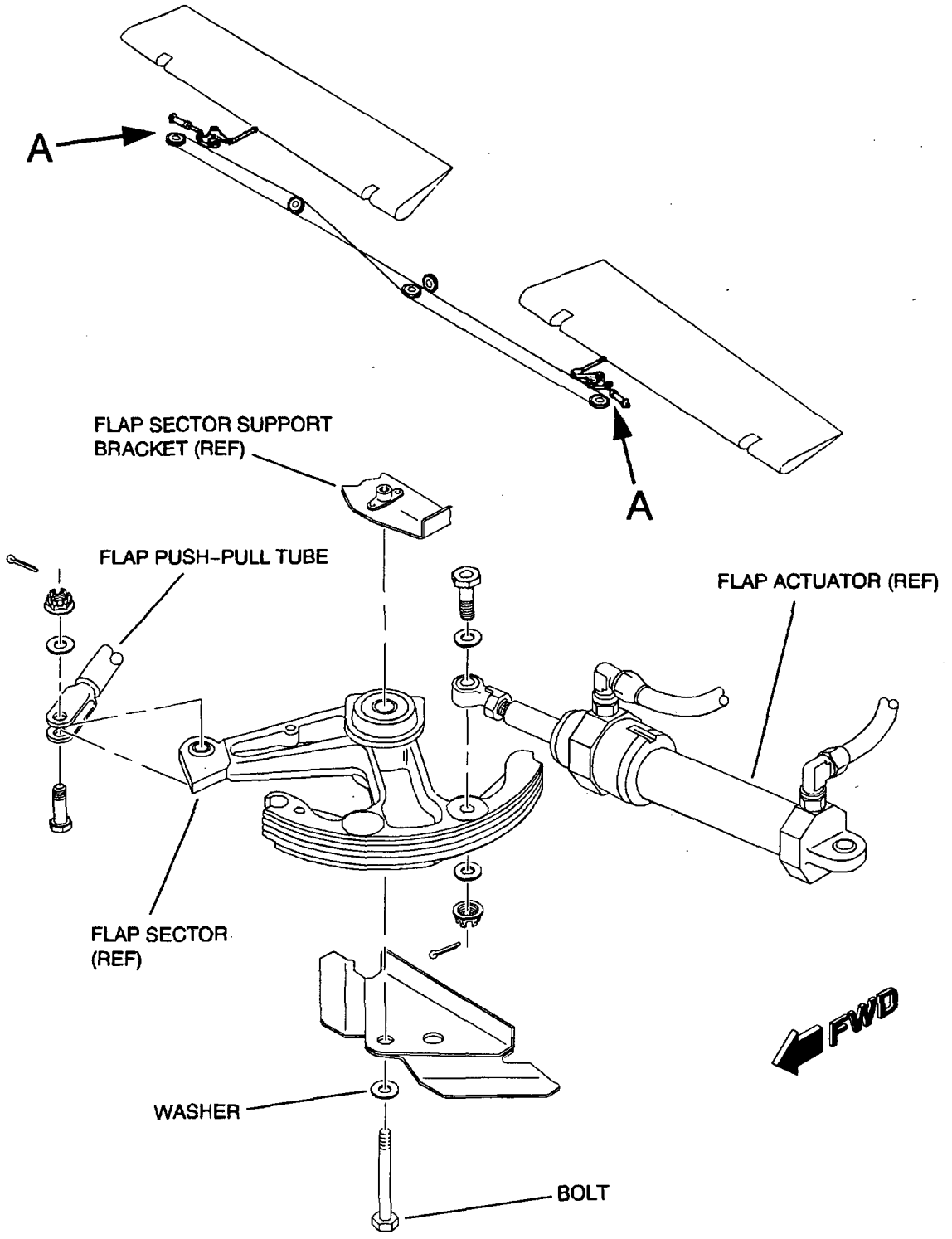
B. Installation of Flap Sector (See Figure 201.)

(1) Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Safety Wire	MS20995C32	Commercially Available	To safety sector bolt.

(2) Prior to installation of flap sector, check all components for excessive wear.
(3) Install flap sector with existing hardware and safety bolt.
(4) Connect flap actuator to flap sector. (Refer to 27-51-07, Removal/Installation.)
(5) Connect push-pull tube to flap sector. (Refer to 27-51-01, Removal/Installation.)
(6) Perform Flap System Functional Test. (Refer to 27-51-00, Adjustment/Test.)
(7) Install flap sector access covers on LH and RH wings.
(8) Disconnect external hydraulic power source from aircraft.
(9) Disconnect electrical power source from aircraft.



(LH SHOWN; RH OPPOSITE)

Detail A

M31-275009-201-01

Flap Sector Installation
Figure 201

SPOILERS - DESCRIPTION AND OPERATION

1. Description

A. The spoilers, located on the upper surface of the wing forward of the flaps, when hydraulically actuated, will hinge up at a specified angle. (See figure 1.) The spoilers are used either symmetrically as speed brakes or asymmetrically as augmentation to the ailerons for improved lateral control. The spoilers are controlled manually by a switch on the cockpit center pedestal. Spoilers are also controlled automatically through a computer-amplifier during aileron augmentation mode (spoileron mode). Refer to 27-10-00.

B. The spoiler system consists of two spoiler surfaces, two down-and-locked switches, two spoiler limit switches, two servo valves mounted on a manifold assembly, a spoiler control valve, a solenoid-operated restrictor valve, a 10-micron filter, a computer-amplifier, a followup on each spoiler and aileron, a spoiler warning light and an augmented aileron warning light. On *Aircraft 35-117 and Subsequent and 36-033 and Subsequent*, the solenoid-operated restrictor valve and the spoiler control valve are a part of the manifold assembly.

C. The spoilers augment the aileron system when flaps are extended below 25°. During the augmentation mode the spoilers work independently and are controlled by an aileron monitoring system. Refer to 27-10-00 for additional information.

D. The spoilers are hinged to the wing at two places and attached to the actuator in the center. Adjustable switches adjacent to the actuators shut down aileron augmentation in the event of failure. Two other switches, adjacent to the limit switches, actuate the spoiler warning light when the spoilers are not down and locked. Two 13° limit switches, one adjacent to each flap sector, cause the spoiler warning light to flash when the flaps are extended below 13° with the Spoiler Switch set to EXT.

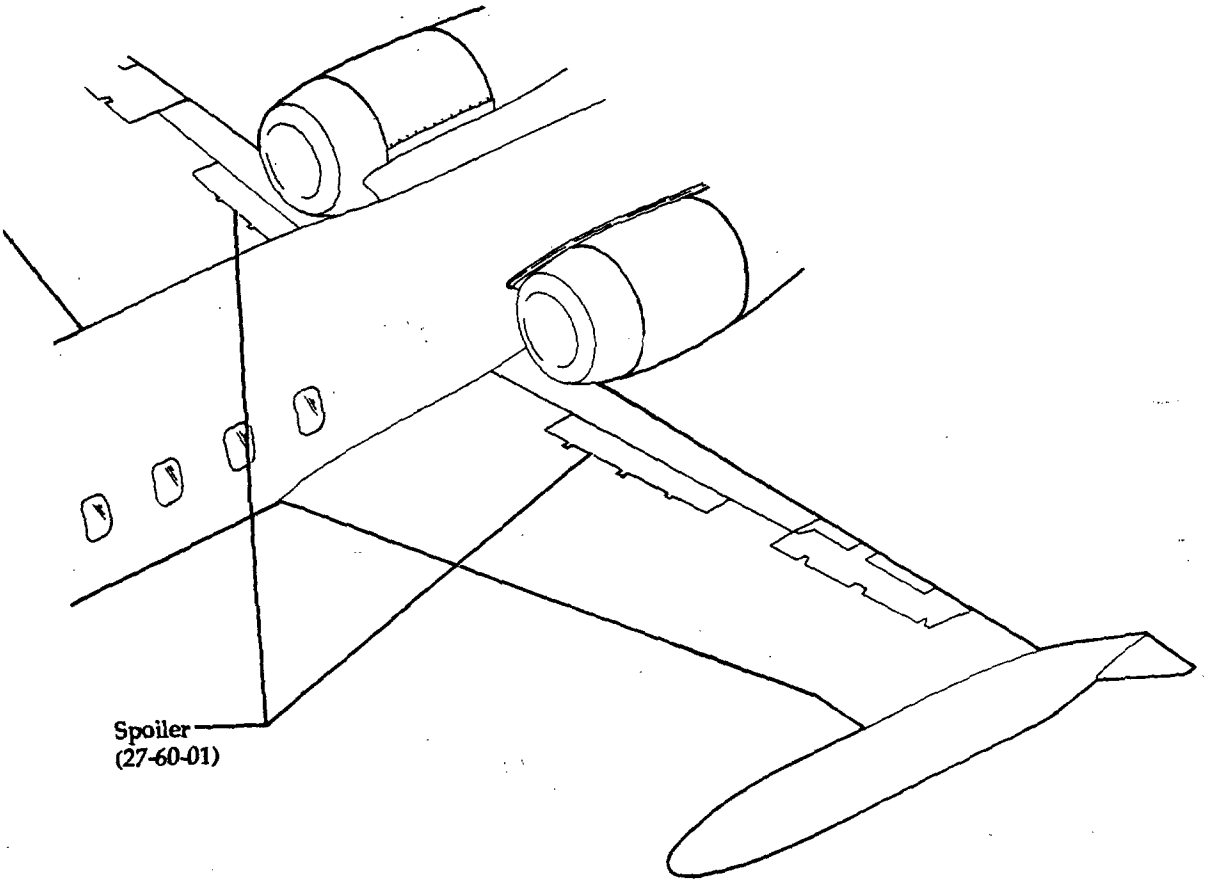
E. On *Aircraft 35-002 thru 35-065, except 35-050, and 36-002 thru 36-017*, the spoileron computer is located beneath the floorboards at FS 332 and LBL 10. On *Aircraft 35-050, 35-066 thru 35-296 and 36-018 thru 36-044*, the spoileron computer is located at FS 286 and LBL 12. On *Aircraft 35-297 and Subsequent and 36-045 and Subsequent*, the spoileron computer is located at FS 310 and LBL 12. On *Aircraft equipped with camera installation*, the spoileron computer is located at FS 275 and LBL 12. (See figure 1.)

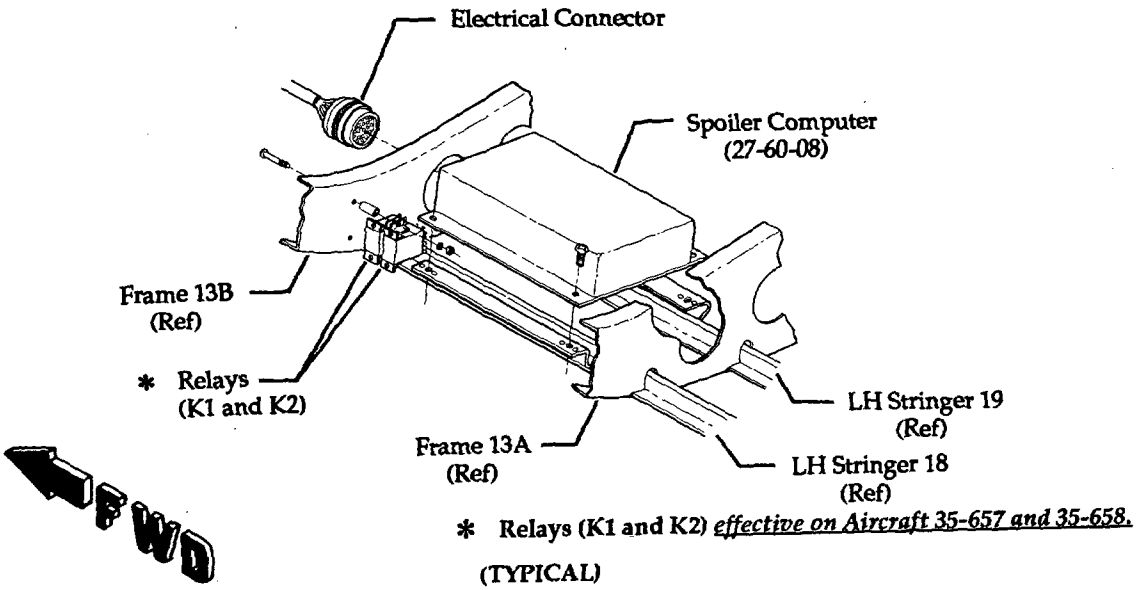
F. Two servo valves mounted on a manifold assembly, are located on the forward side of wing spar 8 in the right wing. On *Aircraft 35-002 thru 35-116 and 36-002 thru 36-032*, the spoiler control valve is located on the upper wing skin between wing spars 7 and 8 and just outboard of the manifold assembly. On *Aircraft 35-117 and Subsequent and 36-033 and Subsequent*, the spoiler control valve is mounted on, and ported directly into, the manifold assembly.

G. On *Aircraft 35-657 and 35-658*, two relays (K1 and K2) are located on the aft side of frame 13B, at LH stringer 18. The relays provide a ground signal to the spoileron computer when either flap is between 0° and 13°. (See figure 1.)

2. Operation (See figures 2 and 3.)

A. Whenever the Spoiler Switch is set to RET, the spoilers will automatically engage to augment the ailerons when the flaps are extended beyond 25°. The spoiler computer-amplifier monitors aileron movement during aileron augmentation. As the ailerons move, the computer-amplifier actuates the spoiler control valve and the spoiler servo valves. As one aileron moves up, the servo valves are actuated to the applicable position and the spoiler on the same wing as the raised aileron extends while the opposite spoiler remains streamlined. The spoiler will rise at a 1:1 ratio with the aileron through 14.5°. A limit switch adjacent to each spoiler actuator limits spoiler extension to 16° (+1°; -0°) during aileron augmentation. A monitor circuit automatically disengages the augmentation system, illuminates the amber AUG AIL light on the glareshield and retracts the spoilers if a system malfunction occurs. Depressing the SPOILERON RESET Switch will restore system operation if the malfunction has cleared itself. The augmentation mode disengages automatically when flaps are retracted above 25° or is





* Relays (K1 and K2) *effective on Aircraft 35-657 and 35-658.*

(TYPICAL)

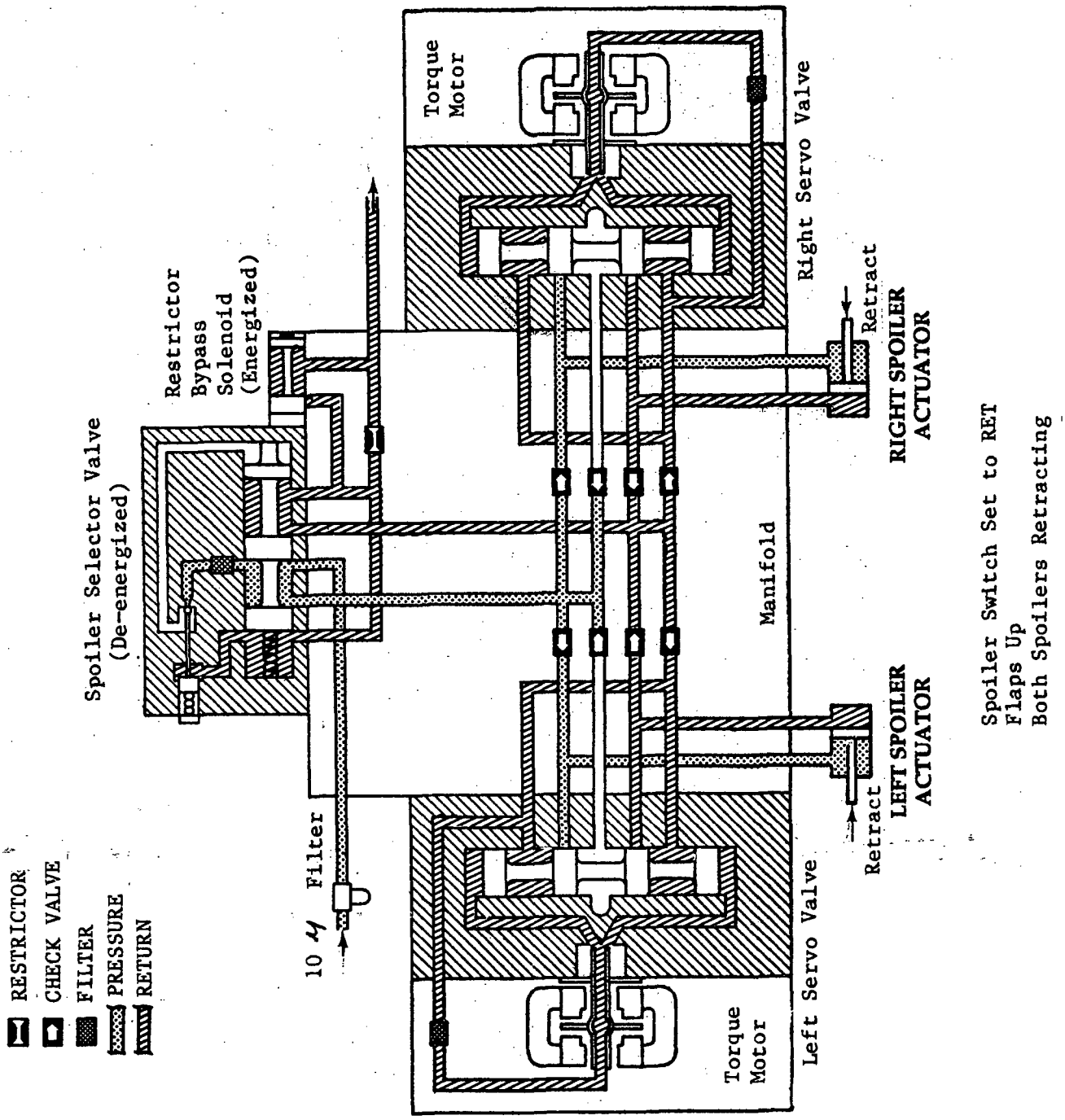
Detail A

Spoiler Locator
Figure 1

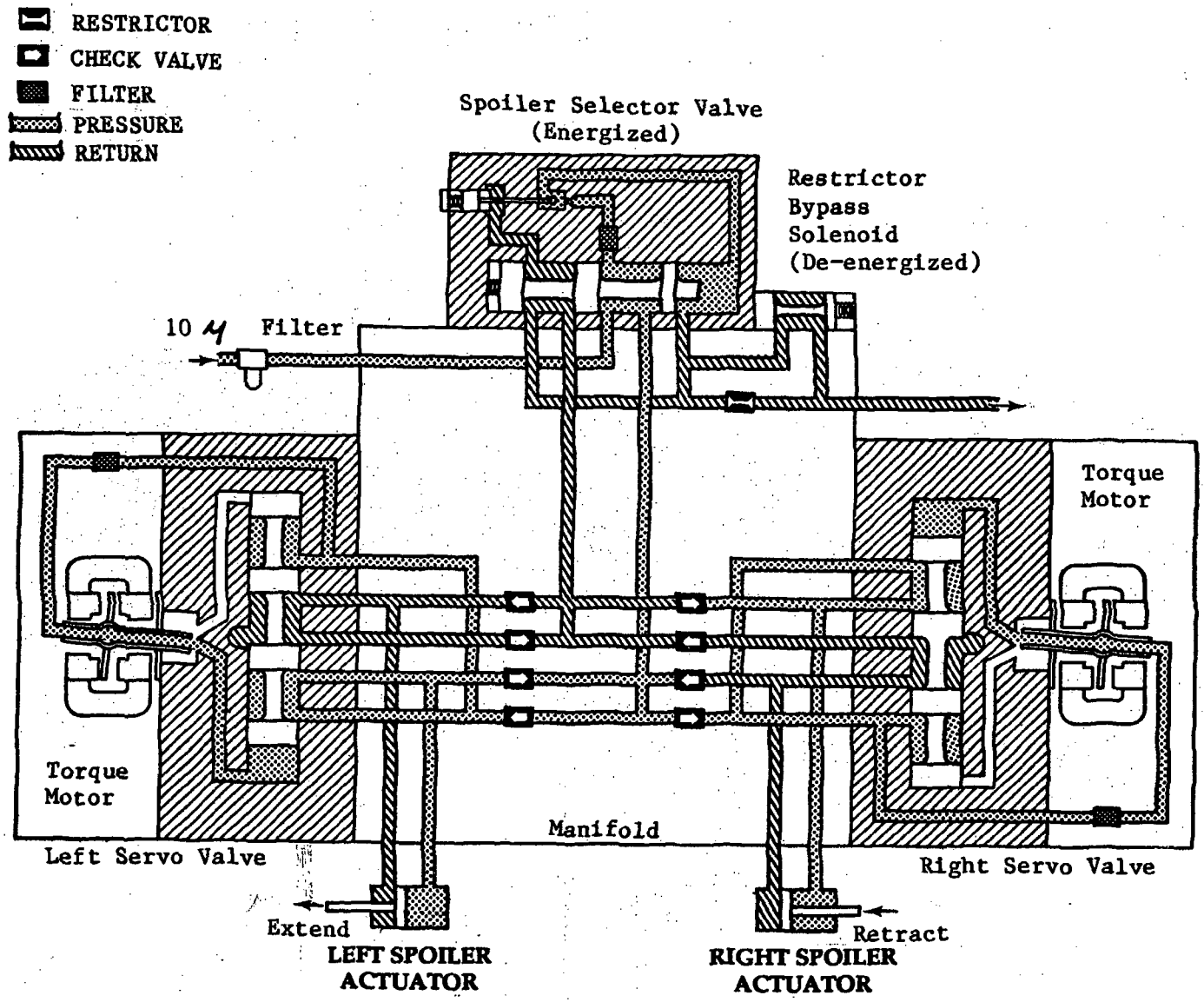
A9-219B

EFFECTIVITY: ALL

overridden by the Spoiler Switch. Normal spoiler extension and retraction is accomplished by setting the Spoiler Switch, on the cockpit center pedestal, to EXT or RET. Setting the switch to EXT applies 28 vdc to the computer-amplifier. The computer then disengages the aileron augmentation mode, if it is being used, and commands both servo valves to the spoiler extend position. At this same time, the computer also actuates the spoiler control valve to the extend position. Hydraulic pressure is then directed through the filter, the spoiler control valve and through the manifold to the extend ports of the spoiler actuators. As the spoilers begin to rise, and as soon as either spoiler rises above 1°, the red SPOILER light on the glareshield will illuminate. When both 1° limit switches have been actuated, the computer will energize the solenoid which closes the restrictor bypass and switch the flow restrictor into the return line. The spoilers will require approximately four seconds to completely extend. Spoiler extension is limited by the actuator stroke. Setting the Spoiler Switch to RET removes the 28 vdc from the computer. The computer then commands the spoiler control valve to the retract position and closes the servo valves. Hydraulic pressure is then directed through the filter, the spoiler control valve, and the manifold to the retract port of the actuators. The restrictor solenoid is de-energized and the restrictor bypass is opened after the spoilers have retracted.

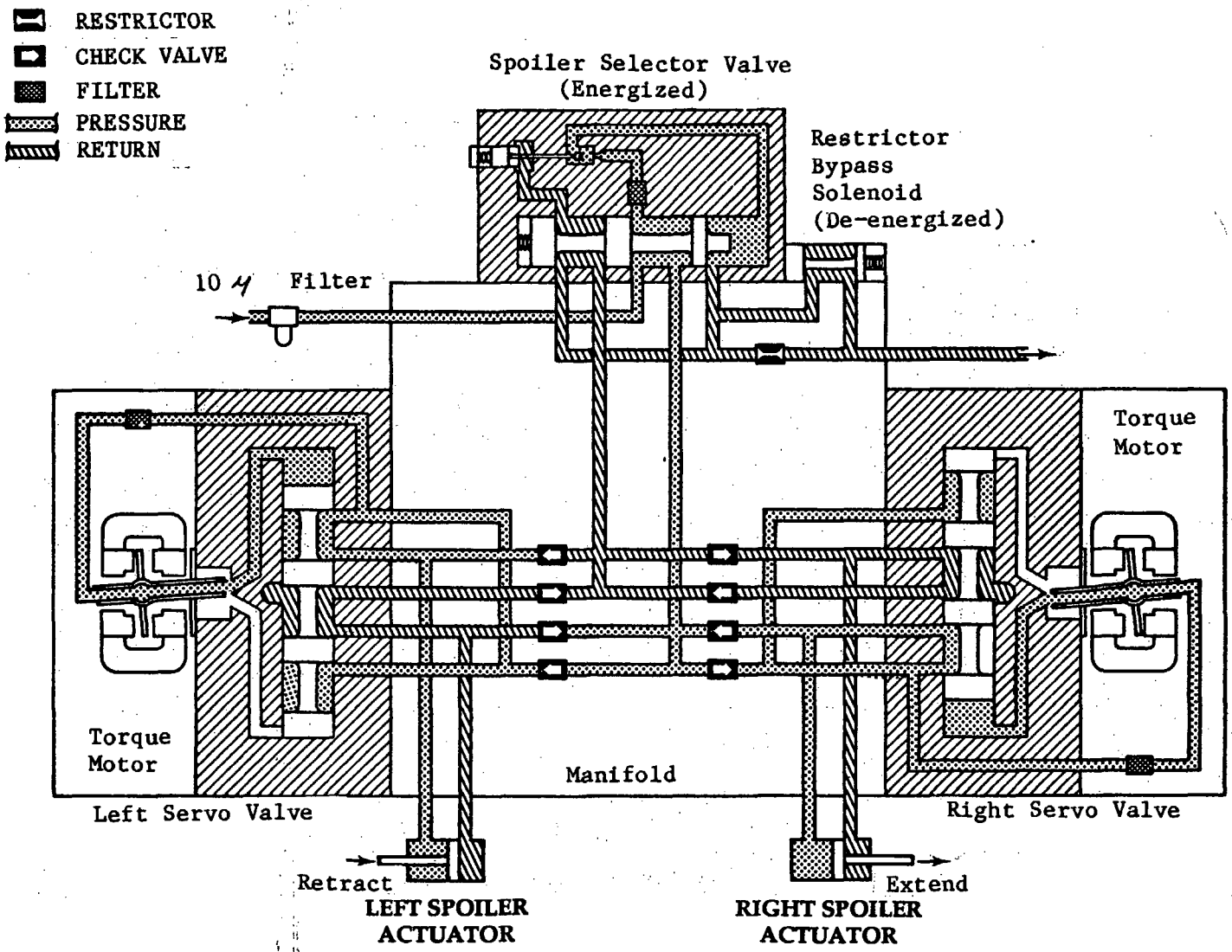


Spoiler System Hydraulic Schematic
Figure 2 (Sheet 1 of 4)



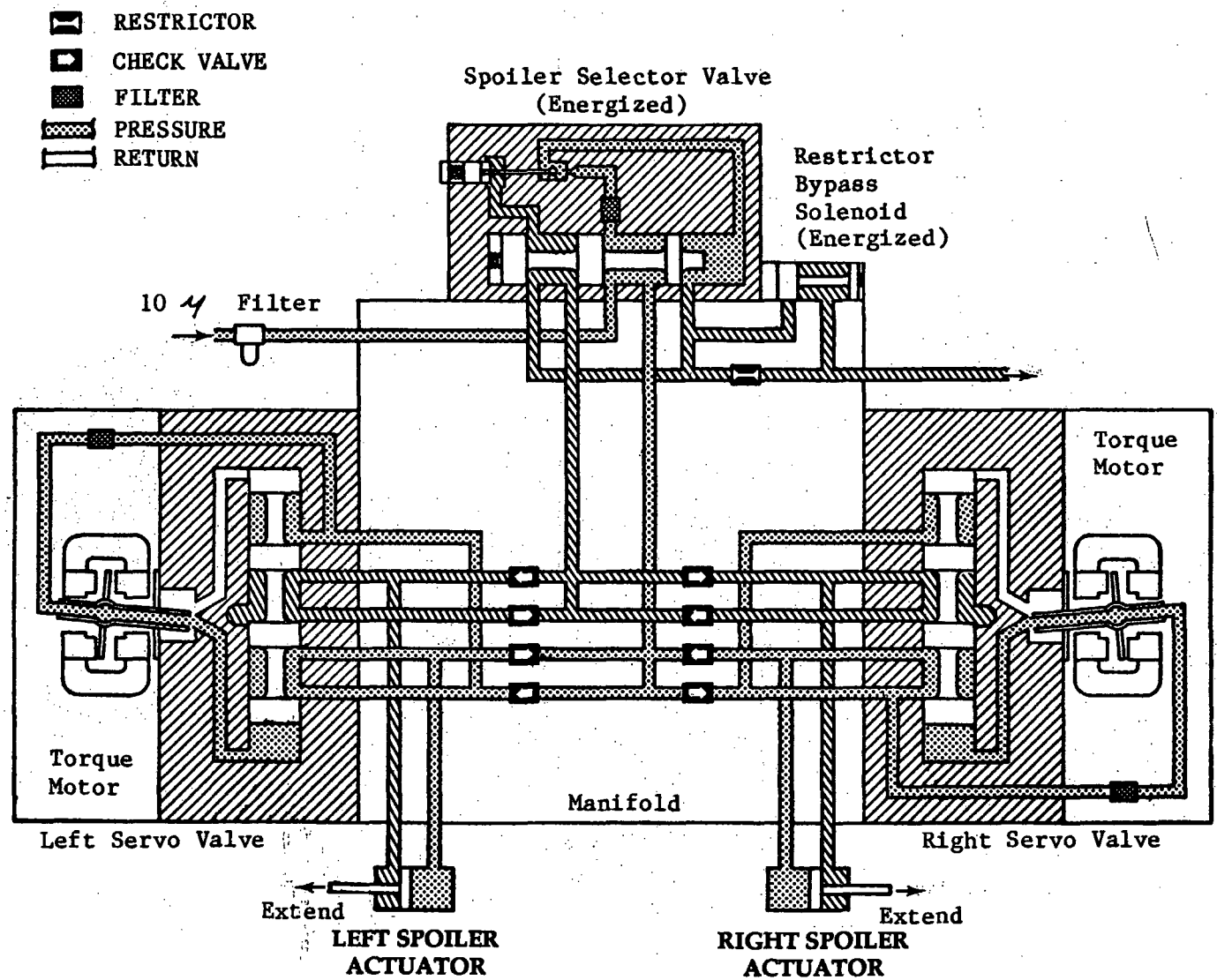
Spoiler System Hydraulic Schematic
Figure 2 (Sheet 2 of 4)

Spoiler Switch Set to RET
Flaps Extended Below 25°
Left Spoiler Rising - Left Aileron Augmented



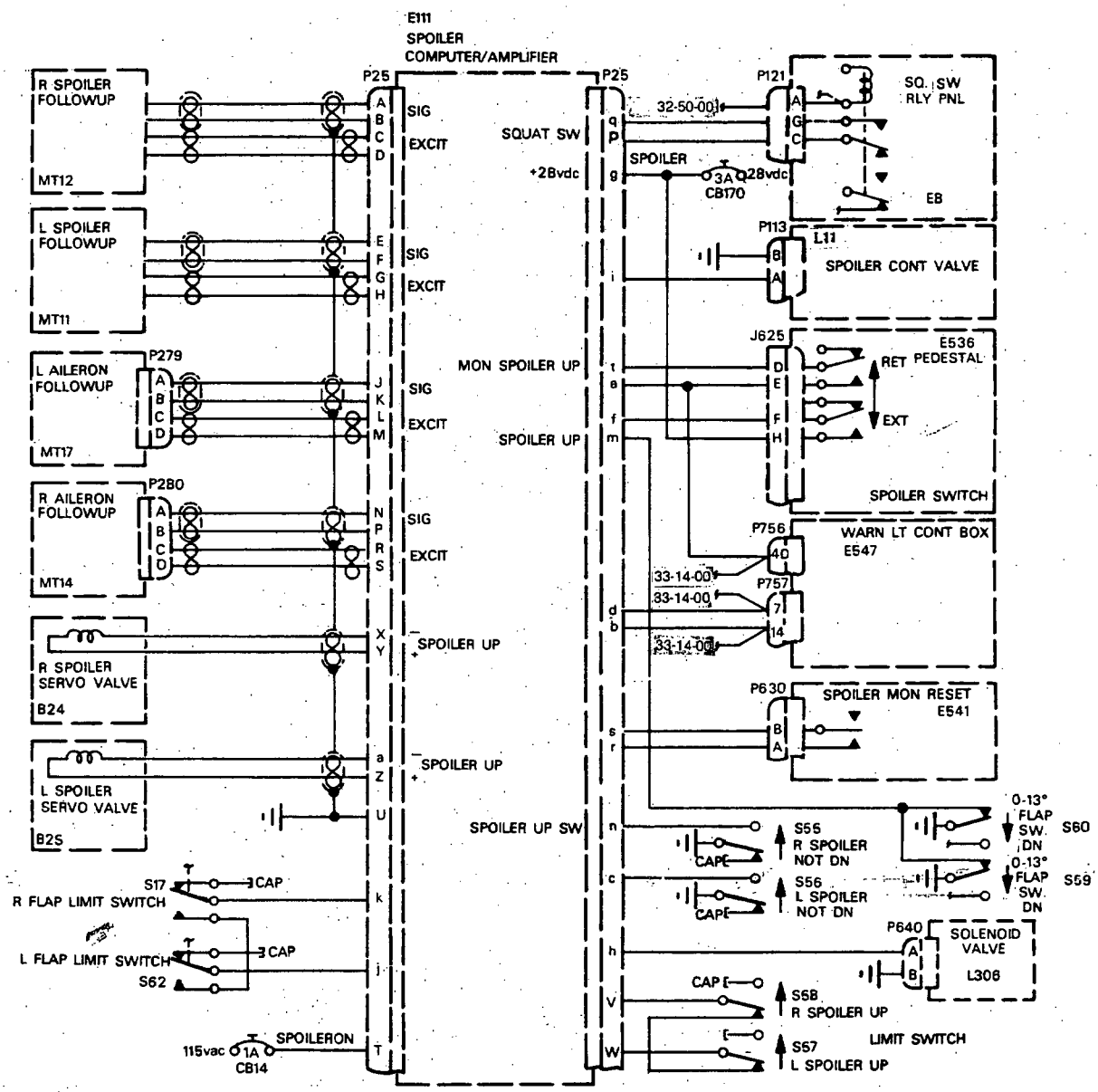
Spoiler System Hydraulic Schematic
Figure 2 (Sheet 3 of 4)

Spoiler Switch Set to RET
Flaps Extended Below 25°
Right Spoiler Rising - Right Aileron Augmented

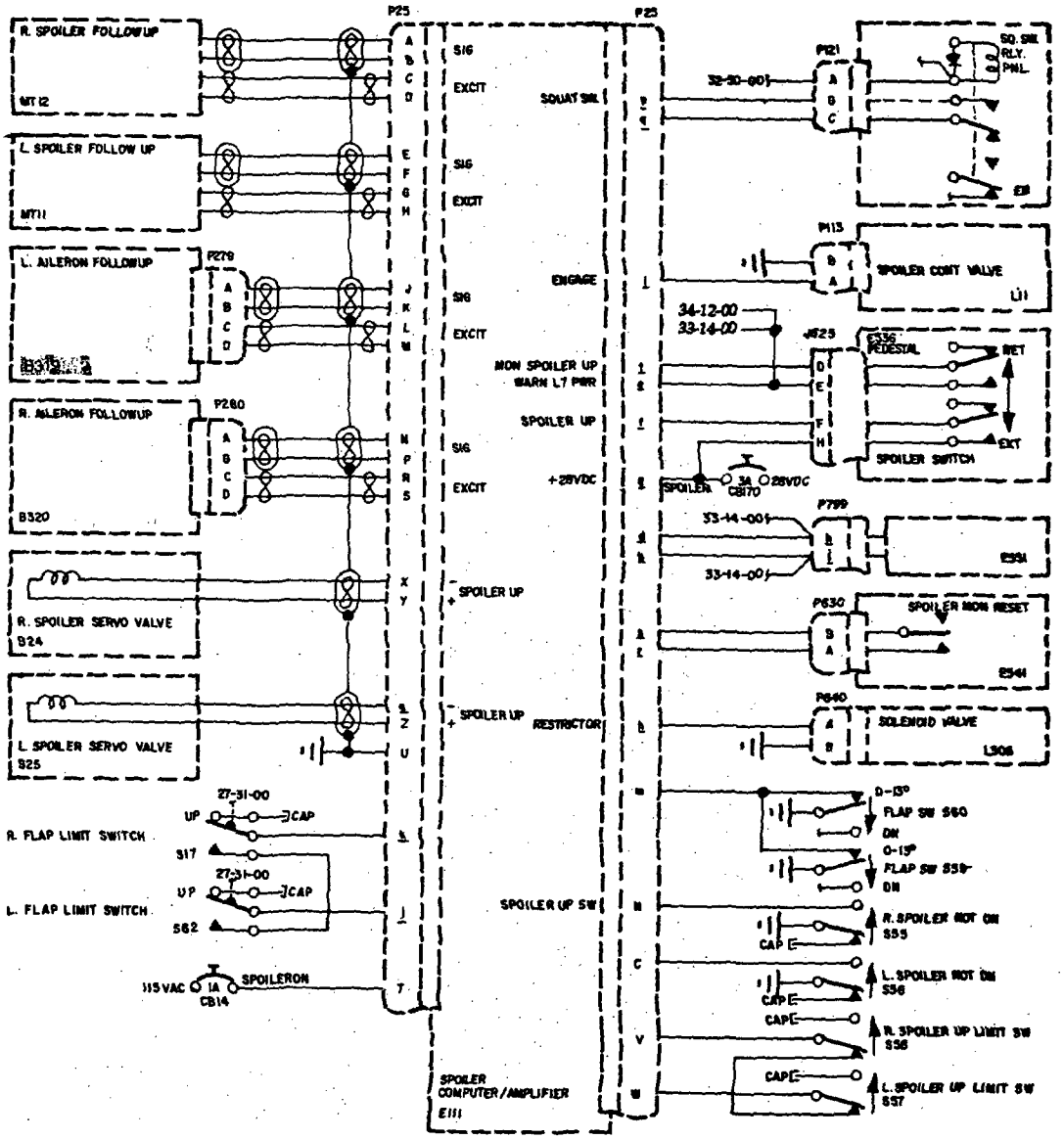


Spoiler System Hydraulic Schematic
Figure 2 (Sheet 4 of 4)

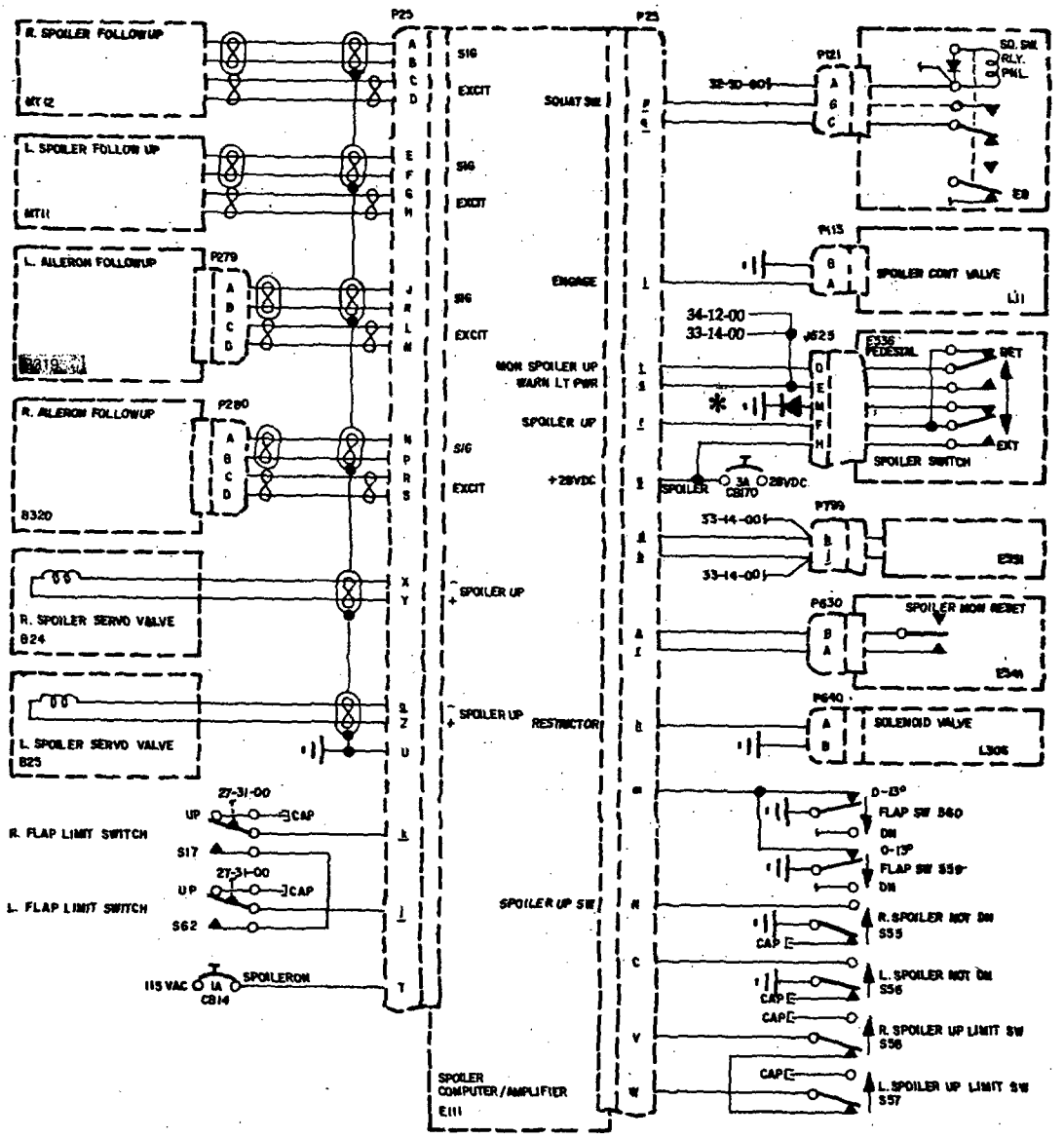
Spoiler Switch Set to EXT
Both Spoilers Extending



Spoiler System Electrical Control Schematic
Figure 3 (Sheet 1 of 9)

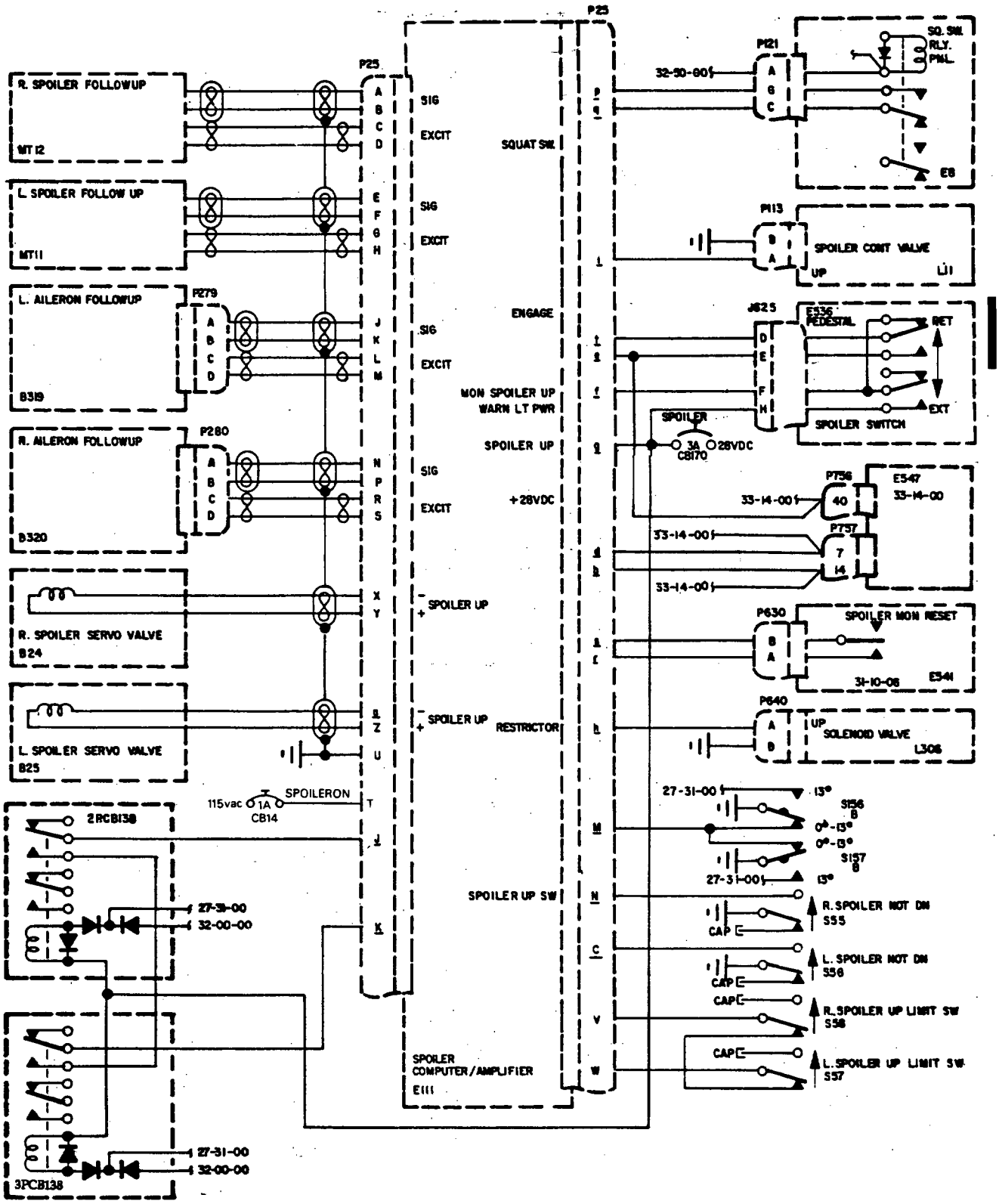


Spoiler System Electrical Control Schematic
Figure 3 (Sheet 2 of 9)

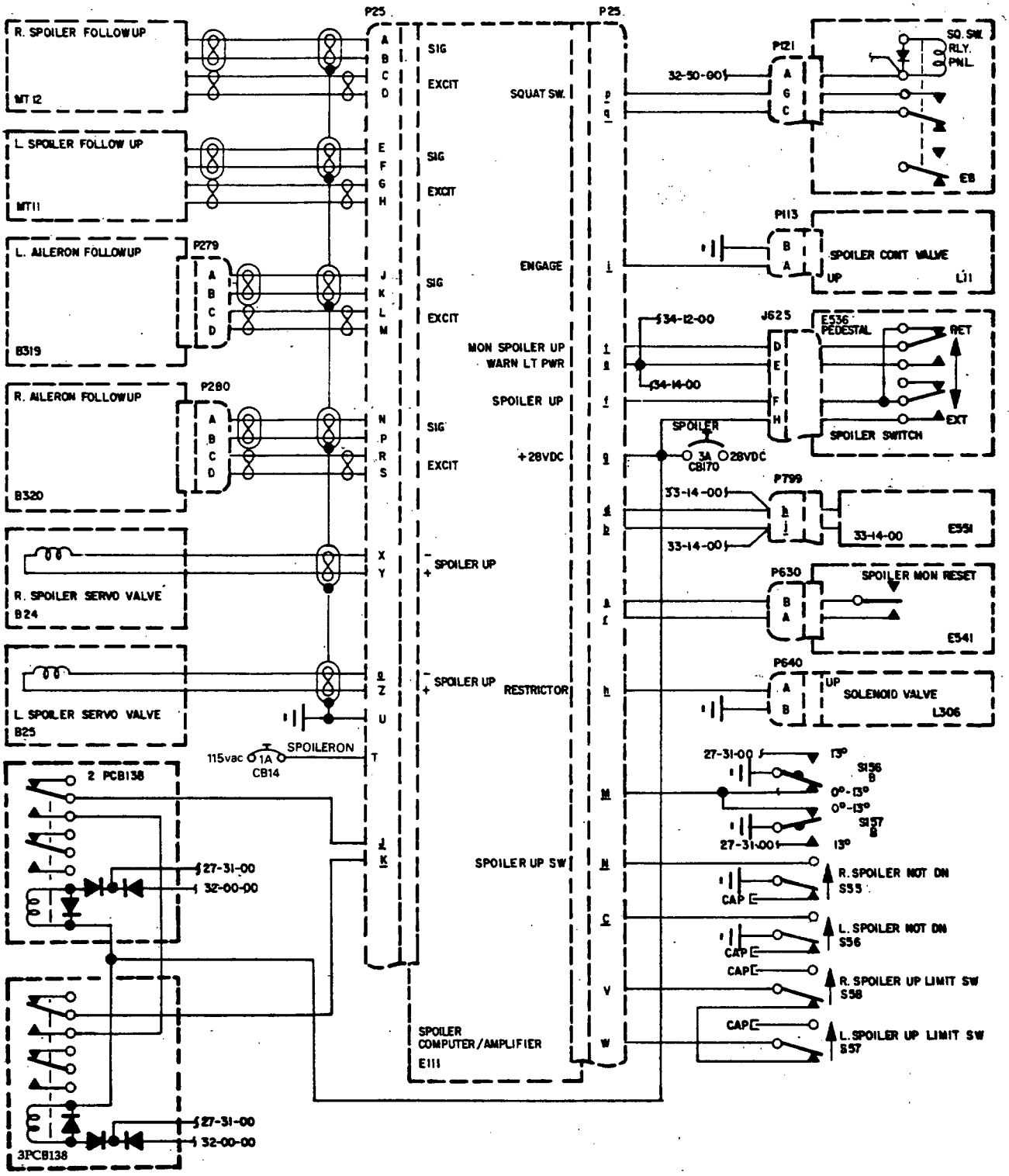


* **Diode effective on *Aircraft 35-040, 35-043, 35-052 thru 35-066, and 36-017.***

Spoiler System Electrical Control Schematic
Figure 3 (Sheet 3 of 9)

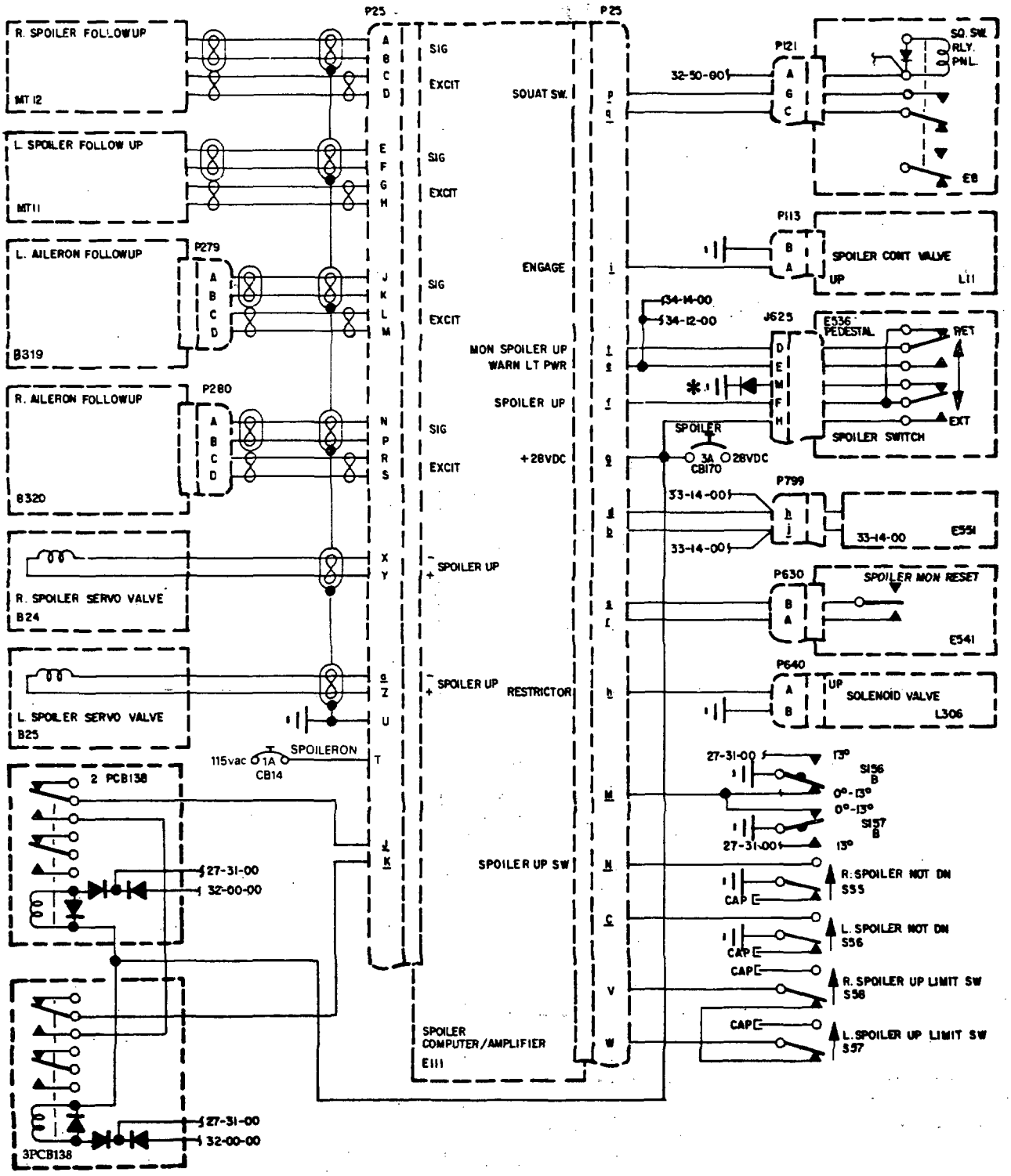


Spoiler System Electrical Control Schematic
Figure 3 (Sheet 4 of 9)



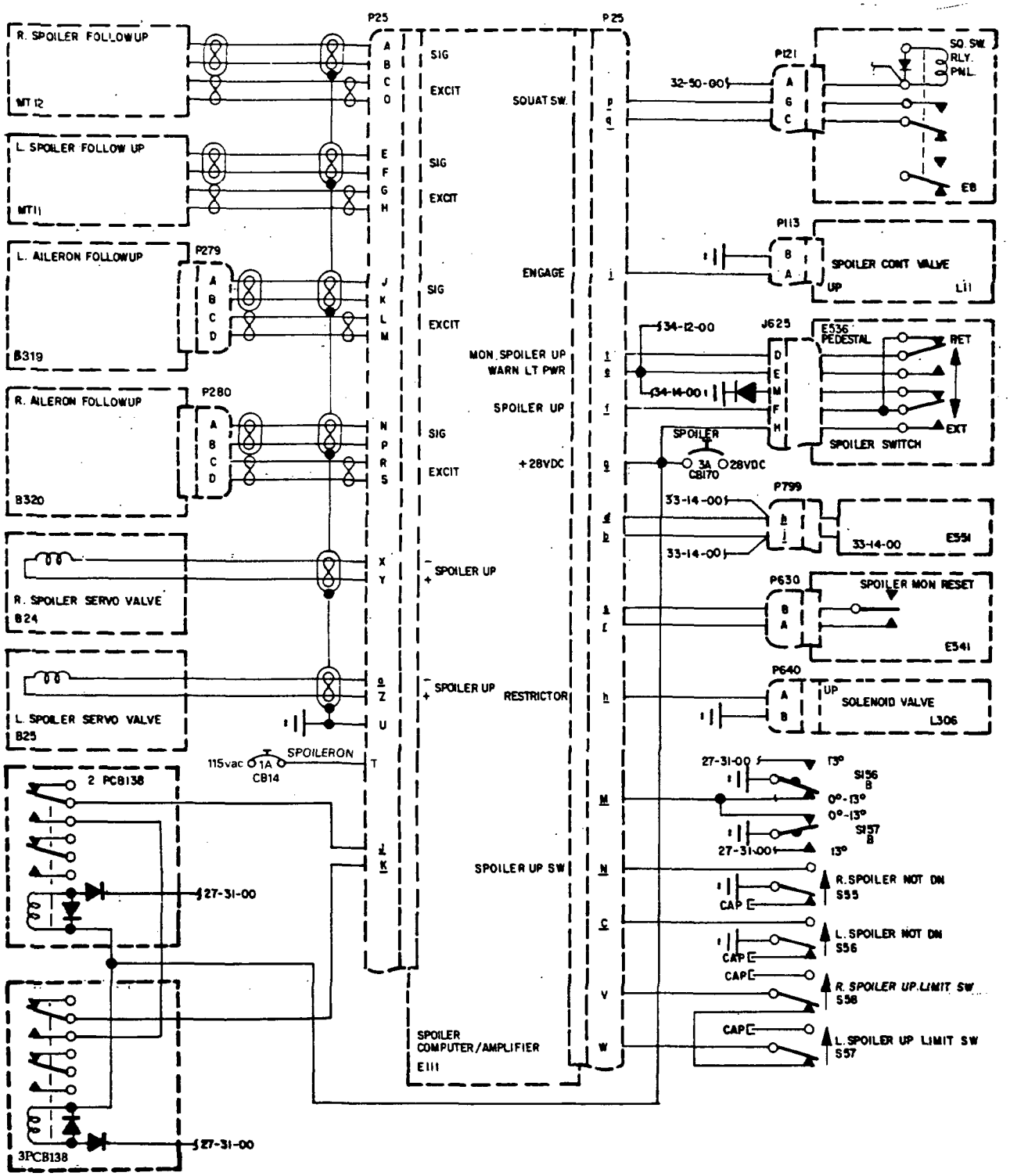
Spoiler System Electrical Control Schematic
Figure 3 (Sheet 5 of 9)

EFFECTIVITY: 35-005 THRU 35-036, 35-041, 35-042, 35-044 THRU 35-051, AND 36-003 THRU 36-013, 36-015, 36-016 EQUIPPED WITH REDUCED APPROACH SPEED SYSTEM

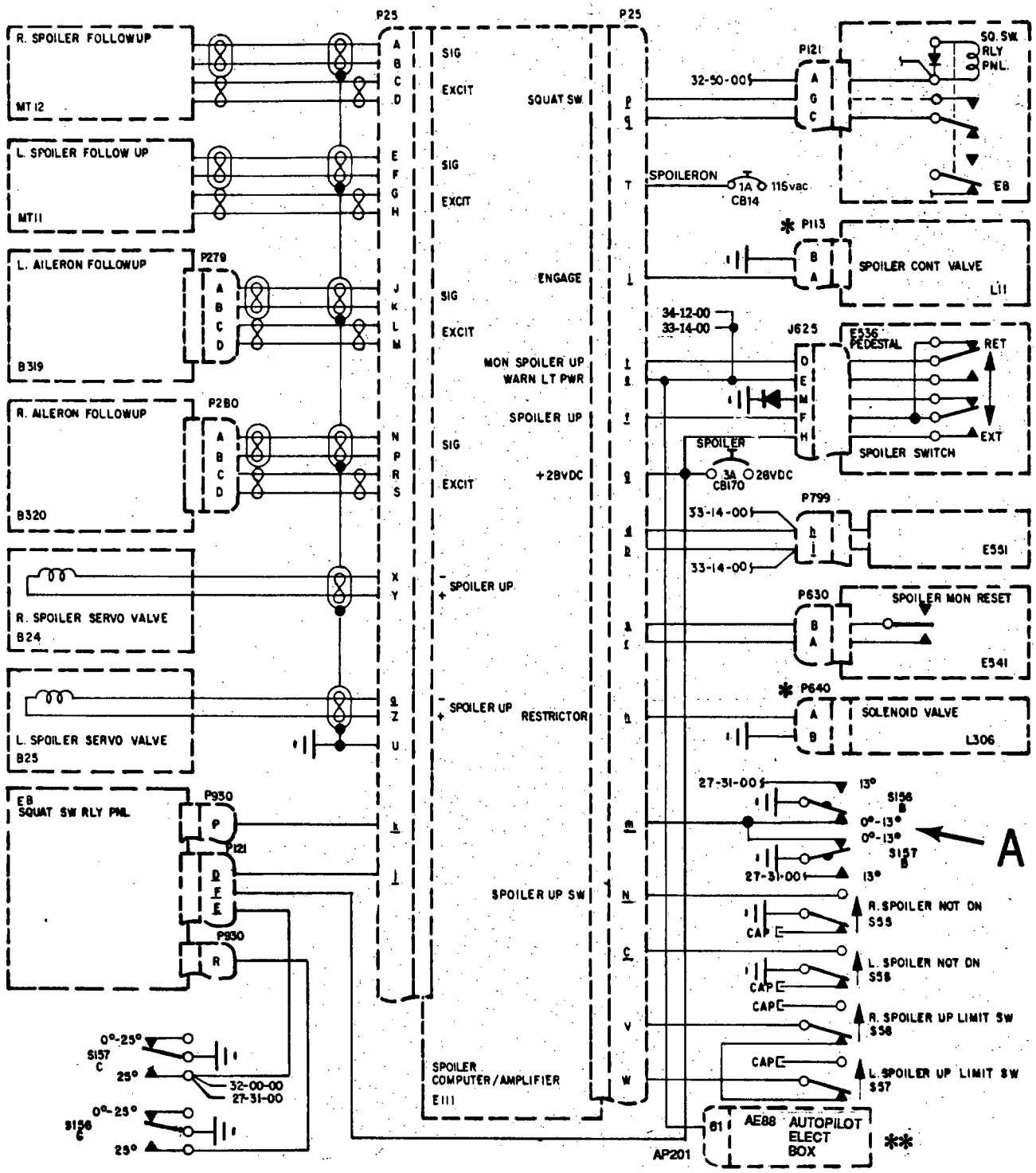


* Diode effective on *Aircraft 35-040, 35-043, 35-052 thru 35-059, and 36-017.*

Spoiler System Electrical Control Schematic
Figure 3 (Sheet 6 of 9)



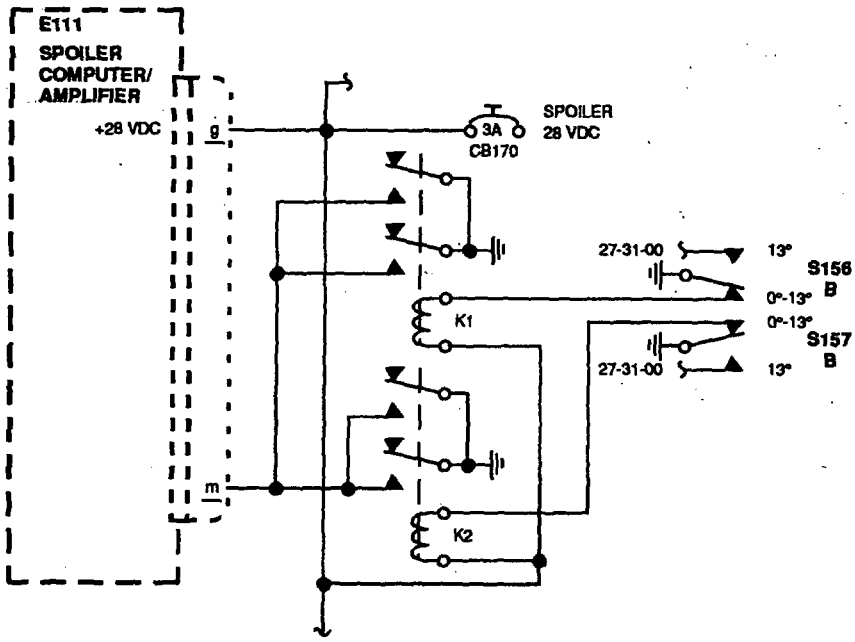
Spoiler System Electrical Control Schematic
Figure 3 (Sheet 7 of 9)



* *Effective Aircraft 35-117 and Subsequent, 36-033 and Subsequent,* P113-B changes to P113C, P113A changes to P113D, P640-A changes to P640-D, and P640-B changes to P640-C.

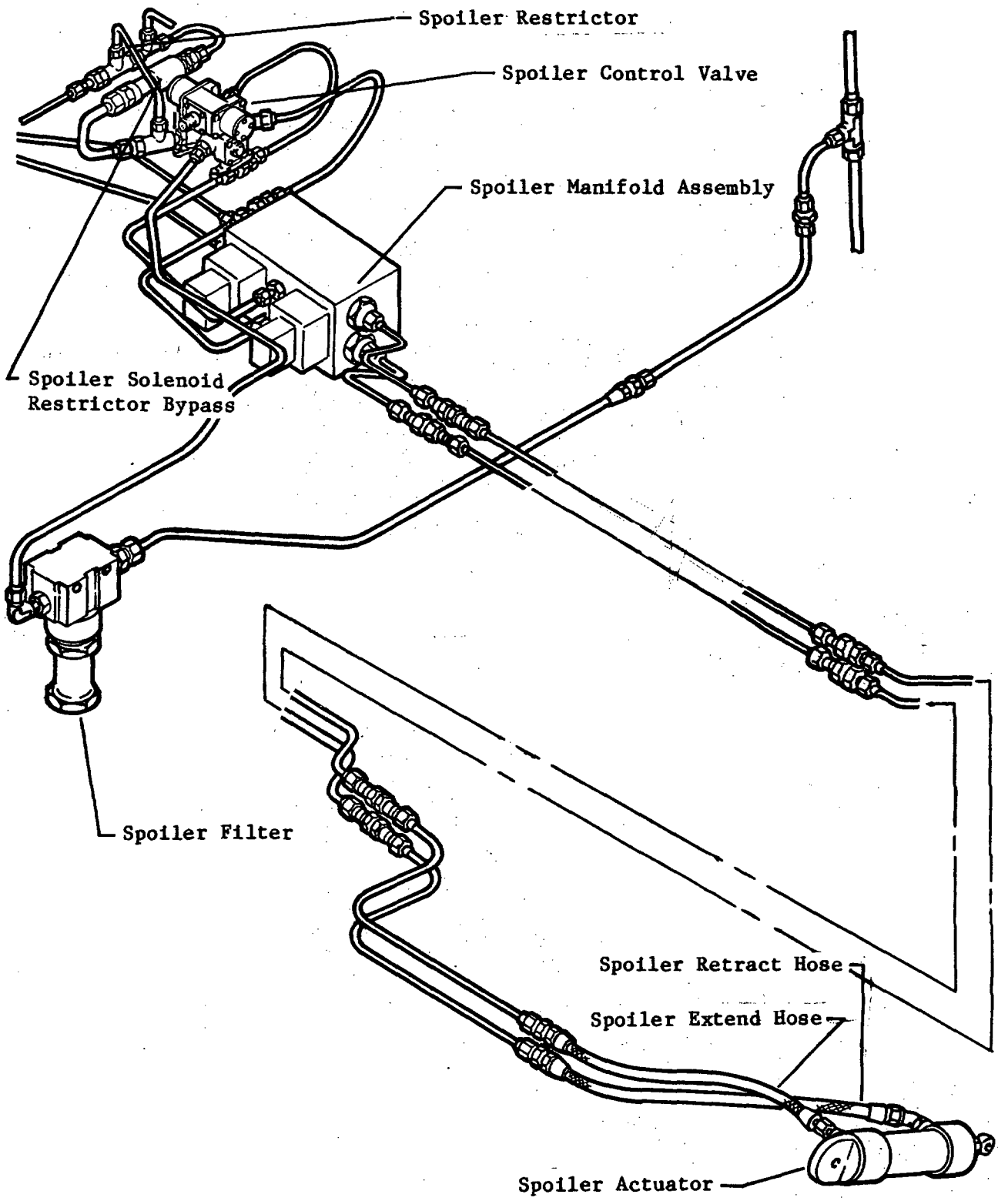
** Autopilot Electrical Box *effective Aircraft 35-506 and Subsequent, and 36-054 and Subsequent.*

Spoiler System Electrical Control Schematic
Figure 3 (Sheet 8 of 9)

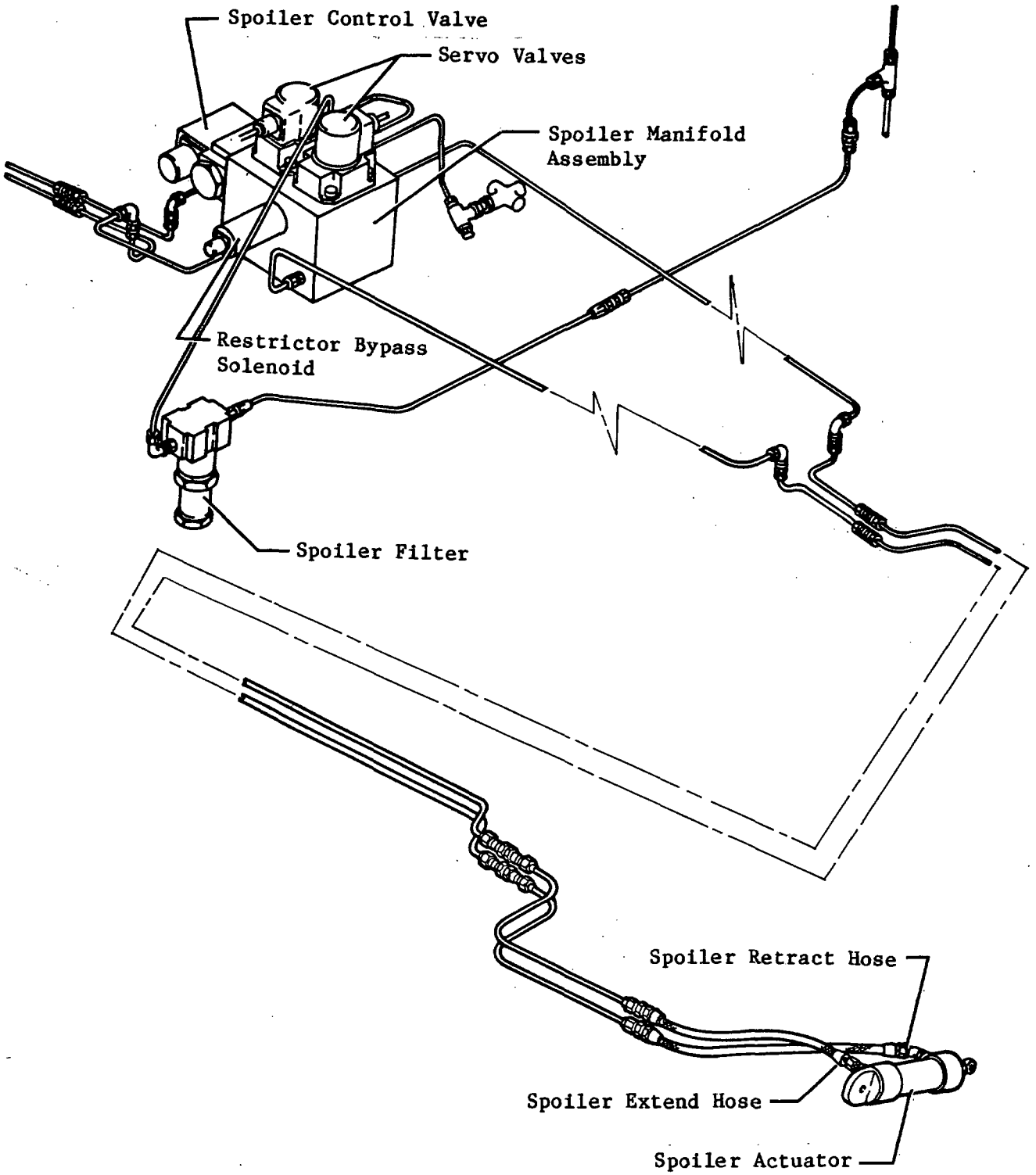


Detail A

Spoiler System Electrical Control Schematic
Figure 3 (Sheet 9 of 9)



Spoiler Hydraulic Components
Figure 4 (Sheet 1 of 2)



Spoiler Hydraulic Components
Figure 4 (Sheet 2 of 2)

SPOILER SYSTEM - ADJUSTMENT/TEST

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Vernier Clinometer	TB107	Learjet Inc. Wichita, KS	To measure angles of control surfaces.
Digital Protractor (DP-45) (Alternate)	02475-01	Lucas & Schaebitz Pennsauken, NJ	To measure angles of control surfaces.
Hydraulic Ground Power Cart (1500 psi max. and flow of 5 gpm)		Commercially Available	Hydraulic power.
Electrical Power Unit (28 vdc)		Commercially Available	Electrical power.
Spoiler Cage	61341	Learjet Inc. Wichita, KS	Safety equipment.
Phase Detector/ Test Box	2671007-6	Learjet Inc. Wichita, KS	Test spoiler phase.
AC Voltmeter (VTVM)		Commercially Available	Voltage measurement.
Rigging Pin		Local Manufacture	Rigging.

2. Adjustment/Test

NOTE: Spoiler rigging is to be accomplished after spoiler or spoiler actuator replacement, limit and down-and-locked switch replacement or any time spoiler actuator rod end adjustment is required.

Spoiler rigging consists of adjusting the spoilers to streamline, synchronizing both spoilers, adjusting spoiler follow-ups, aileron follow-ups, down-and-locked switch, limit switch and adjusting spoiler computer potentiometers.

Prior to rigging the spoilers, ensure that ailerons are properly rigged. (Refer to Adjustment/ Test, 27-10-00.)

A. Rig Spoilers (See Figure 201.)

WARNING: PLACE CAGES OVER SPOILERS WHEN RIGGING. DURING PROCEDURES THE SPOILERS CAN SLAM DOWN CAUSING BODILY INJURY.

NOTE: Slam down is a quick spoiler retraction (less than one second). It is markedly different from the normal retraction time.

(1) Ensure that spoiler actuator is secured to wing attach bracket and that actuator rod end is screwed well out of actuator to avoid spoiler or wing damage.
(2) Lower tailcone access door and connect hydraulic power unit to aircraft hydraulic quick-disconnects.
(3) Apply 28 vdc to the aircraft and turn on inverter.
(4) Cycle spoilers using Spoiler Switch. Check that spoilers fair to wing when retracted. If necessary, loosen jamnut on actuator rod end and remove attaching bolt on the actuator control arm assembly and adjust actuator rod end until spoiler becomes faired with wing.
(5) Pull SQUAT SW circuit breaker to simulate flight condition.
(6) Check spoiler extension and retraction time. On *Aircraft 35-002 thru 35-505, 36-002 thru 36-053*, maximum time shall be 3.8 seconds or less.
(7) On *Aircraft 35-002 thru 35-505, 36-002 thru 36-053, modified per AAK 82-3, 35-506 and Subsequent, 36-054 and Subsequent*, maximum time shall be 5 to 7 seconds. Extension time can be adjusted using SLO UP potentiometer on front of spoiler computer.
(8) Place clinometer on spoiler surface and set clinometer to zero.
(9) Cycle spoilers and check for full extension of 47 (+0/-7)°. Spoilers must be synchronized within a maximum travel differential of 1°.
(10) Extend spoilers. Remove safety wire and loosen jamnuts on spoiler down-and-locked switches. Adjust jamnuts until LH and RH switches actuate against spoiler rub block and spoiler warning light will extinguish. Adjust jamnuts one revolution beyond point of actuation. Secure and safety wire jamnuts.
(11) Turn off inverter and remove electrical and hydraulic power from the aircraft.
(12) Remove actuator attach bolt from spoiler control arm and remove actuator from control arm.

NOTE: Rigging pin is fabricated from 3/16 inch (5 millimeter) round steel stock. Make 90° bend, approximately 8-1/2 inches (215 millimeters) from end of 12 inch (305 millimeter) piece of stock. Paint short end of piece red (for visibility).

(13) Center the ailerons to neutral. Insert rigging pin in aileron sector.
(14) Extend flaps beyond 25°.
(15) Gain access to the spoiler computer, disconnect spoiler computer connector (P25) and connect phase detector/test box (P/N 2671007-6) between aircraft wiring and spoiler computer.
(16) Apply 28 vdc to the aircraft and turn on inverter. Do not apply hydraulic power.
(17) Ground AC voltmeter between right spoiler VOLTS terminal and GROUND terminal on the test box.
(18) Loosen screws and rotate right spoiler follow-up until 0.8 volts is obtained and the green RIGHT SPOILER FU light illuminates.
(19) Manually lift right spoiler. The green RIGHT SPOILER FU light will extinguish and the red light will illuminate. If lights illuminate otherwise, the follow-up must be rotated 180°.
(20) Secure right spoiler follow-up.
(21) Disconnect AC voltmeter from right spoiler VOLTS terminal and plug into right aileron VOLTS terminal.

CAUTION: THE AILERON FOLLOW-UP HAS A MECHANICAL STOP. DO NOT FORCE FOLLOW-UP ROTATION THROUGH STOP.

(22) With aileron in neutral, loosen screws and rotate right aileron follow-up (B320) until a minimum voltage null is obtained.
(23) Remove rigging pin from aileron sector and raise right aileron. The green RIGHT AILERON FU light shall illuminate. If the red light illuminates, the follow-up must be rotated 180°.

(24) Secure right aileron follow-up (B320).
(25) Disconnect AC voltmeter from right aileron VOLTS terminal and plug into left spoiler VOLTS terminal.
(26) Loosen screws and rotate left spoiler follow-up until 0.8 volts is obtained and the green LEFT SPOILER FU light illuminates.
(27) Manually lift left spoiler. The green LEFT SPOILER FU light will remain illuminated. If lights illuminate otherwise, the follow-up must be rotated 180°. Do not force follow-up rotation through stop.
(28) Secure left spoiler follow-up.
(29) Disconnect AC voltmeter from left spoiler VOLTS terminal and plug into left aileron VOLTS terminal.

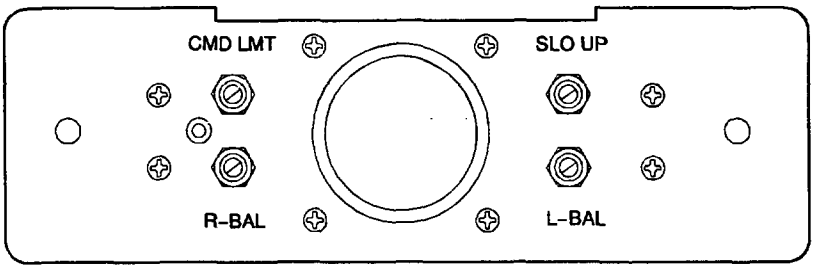
CAUTION: THE AILERON FOLLOW-UP HAS A MECHANICAL STOP. DO NOT FORCE FOLLOW-UP ROTATION THROUGH STOP.

(30) With aileron in neutral (rigging pin in aileron sector), loosen screws and rotate left aileron follow-up (B319) until a minimum voltage is obtained.
(31) Remove rigging pin from aileron sector and raise left aileron. Green LEFT AILERON FU light shall illuminate. If the red light illuminates, the follow-up must be rotated 180°. Do not force follow-up rotation through stop.
(32) Secure left aileron follow-up (B319).
(33) Disconnect voltmeter from test box. Leave remainder of test hook-up.
(34) Attach clinometer to right spoiler. Raise the spoiler to 16°, loosen jamnuts, and adjust spoiler limit switch so that LIMIT SWITCH light (amber) on test box will go out when spoiler is extended beyond 16°. Secure jamnuts. Lower spoiler and again lift it to ensure that LIMIT SWITCH light goes out at 16 (+1/-0)° extension.
(35) Attach clinometer to left spoiler. Raise the spoiler to 16°, loosen jamnut, and adjust spoiler limit switch so that LIMIT SWITCH light (amber) on test box will go out when spoiler is extended beyond 16°. Secure jamnuts. Lower spoiler and again lift it to ensure that LIMIT SWITCH light goes out at 16 (+1/-0)° extension.
(36) Turn off inverter and remove 28 vdc from aircraft. Remove test box and connect spoiler computer.
(37) Position spoiler actuator in spoiler control arm and secure with bolt.

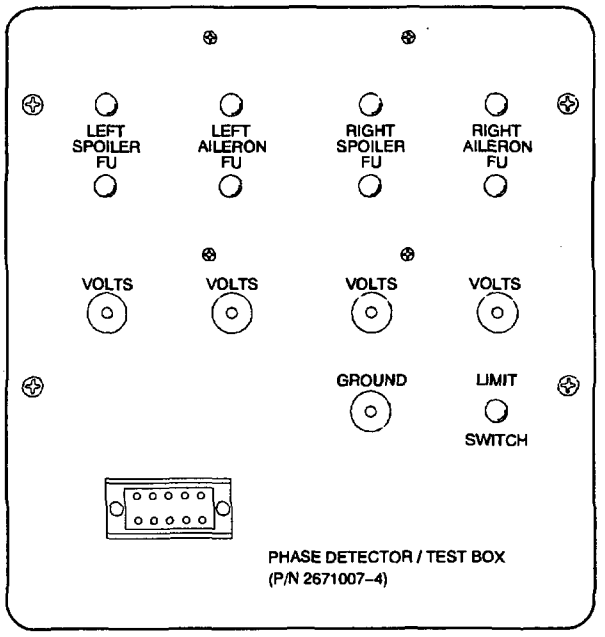
WARNING: PLACE CAGES OVER SPOILERS WHEN RIGGING. THE SPOILERS CAN SLAM DOWN WHEN HYDRAULIC POWER IS APPLIED.

NOTE: Slam down is a quick spoiler retraction (less than one second). It is markedly different from the normal retraction time.

(38) Apply hydraulic power and 28 vdc to aircraft. Turn on inverter.
(39) Center ailerons. Adjust R-BAL potentiometer (R2) on the connector end of the spoiler computer until right spoiler rises approximately 3°. Adjust potentiometer (R2) back until spoiler is in the down (flush with wing) position. Back off potentiometer (R2) slightly below bias down.
(40) With ailerons centered, adjust L-BAL potentiometer (R3) on the connector end of the spoiler computer until left spoiler rises approximately 3°. Adjust potentiometer (R3) until spoiler is in the down (flush with wing) position. Back off the potentiometer (R3) slightly below bias down.
(41) Attach clinometer to left spoiler. Apply full left aileron and adjust the CMD LMT potentiometer (R1) on the connector end of the spoiler computer to obtain 14.5 (±0.5)° spoiler extension.
(42) Attach clinometer to right aileron. Move right aileron to full up then back to 8°. The left spoiler will remain down. The right spoiler will extend 8 (±1)°. If necessary, readjust R-BAL potentiometer (R2) to obtain a 1:1 ratio of aileron and spoiler.



SPOILER COMPUTER



M35-276000-201-01

Spoiler System Rigging
Figure 201

(43) Attach clinometer to left aileron. Move left aileron to full up then back to 8°. The right spoiler will remain down. The left spoiler will extend 8 (±1)°. If necessary, readjust L-BAL potentiometer (R3) to obtain a 1:1 ratio of aileron and spoiler.
(44) Perform Operational Check of Spoilers. (Refer to 27-60-00).

3. Inspection/Check
A. Perform Operational Check of Spoilers

WARNING: PLACE CAGES OVER SPOILERS DURING OPERATIONAL CHECK. THE SPOILERS MAY SLAM DOWN WITHOUT WARNING DURING OPERATIONAL CHECK.

NOTE: Slam down is a quick retraction (less than one second). It is markedly different from normal retraction time.

(1) Lower tailcone access door and connect auxiliary hydraulic power unit to aircraft hydraulic quick-disconnect.
(2) Apply 28 vdc to aircraft and turn on inverter.
(3) Retract flaps and pull SQUAT SW circuit breaker to simulate flight mode.
(4) Set Spoiler Switch to EXT. Spoiler will extend symmetrically within 3°. On *Aircraft 35-002 thru 35-505, 36-002 thru 36-053* extend time shall be 3.8 seconds or less.
(5) On *Aircraft 35-002 thru 35-505, 36-002 thru 36-053 modified per AAK 83-2, 35-506 and Subsequent, 36-054 and Subsequent* extend time shall be 5 to 7 seconds. Spoiler light will illuminate and spoiler extension will be 47 (+0/-7)°. Check spoiler extension with a clinometer.
(6) Set Spoiler Switch to RET. On *Aircraft 35-002 thru 35-505, 36-002 thru 36-053* spoiler will retract in 3.8 seconds or less.
(7) On *Aircraft 35-002 thru 35-505, 36-002 thru 36-053 modified per AAK 82-3, 35-506 and Subsequent, 36-054 and Subsequent,* spoilers will retract in 5 to 7 seconds. Spoilers will be flush with wings. Spoiler lights will extinguish.
(8) Extend flaps below 13°. Set Spoiler Switch to EXT. Spoilers shall extend and spoiler light shall illuminate and flash.
(9) Retract flaps. The spoiler light will cease to flash and remain illuminated. Set Spoiler Switch to RET. Spoiler light will extinguish when spoilers are fully retracted.
(10) Raise RH aileron a few degrees. Begin flap extension while observing the right spoiler. Stop flap extension when the right spoiler begins to move. Check flap extension. Flaps shall be extended 25 (+5/-0)°.
(11) Attach clinometer to RH aileron. Move the right aileron to full up then back to 8°. The left spoiler will remain down and the right spoiler will extend 8 (+1)°.
(12) Attach clinometer to left aileron. Move the left aileron to full up then back to 8°. The right spoiler will remain down and the left spoiler will extend 8 (+1)°.
(13) Move the left aileron to full up. The left spoiler shall be extended 14.5 (±0.5)°. The right spoiler shall not extend.
(14) Move the right aileron to full up. The right spoiler shall be extended 14.5 (±0.5)°. The left spoiler shall not extend.
(15) Alternately and quickly apply full left and full right aileron (turn control wheel as quickly as possible). AUG AIL light shall not illuminate.
(16) Raise and hold the SPOILERON RESET Switch. Rotate control wheel to raise the right aileron. Raise the aileron until AUG AIL light illuminates. Check aileron extension with a clinometer. Extension shall be 4° minimum and 6.5° maximum. Release test switch and return ailerons to neutral.

(17) Raise and hold SPOILERON RESET Switch. Rotate control wheel to raise left aileron. Raise aileron until AUG AIL light illuminates. Check aileron extension with a clinometer. Extension shall be 4° minimum and 6.5° maximum. Release test switch and return ailerons to neutral.

WARNING: PLACE CAGES OVER SPOILERS WHEN RIGGING. DURING PROCEDURES THE SPOILERS CAN SLAM DOWN CAUSING BODILY INJURY.

NOTE: Slam down is a quick spoiler retraction (less than one second). It is markedly different from normal retraction time.

(18) Move the left aileron to full up. The left spoiler will extend. Pull the DC SPOILER circuit breaker. The left spoiler will slam down. Return aileron to neutral and reset SPOILER circuit breaker.
(19) Move the right aileron to full up. The right spoiler will extend. Pull the DC SPOILER circuit breaker. The right spoiler will slam down. Return aileron to neutral and reset SPOILER circuit breaker.
(20) Move the left aileron to full up. The left spoiler will extend. Pull the AC SPOILERON circuit breaker. The left spoiler will slam down. Return aileron to neutral and reset SPOILERON circuit breaker.
(21) Move the right aileron to full up. The right spoiler will extend. Pull the AC SPOILERON circuit breaker. The right spoiler will slam down. Return aileron to neutral and reset SPOILERON circuit breaker.
(22) Pull AC SPOILERON circuit breaker and depress SQUAT SW circuit breaker. Set SPOILER Switch to EXT. Spoilers shall extend in less than 1.0 second. SPOILER light will be flashing. Retract flaps. Check that SPOILER light burns steadily as flaps retract above approximately 13°.
(23) Set SPOILER Switch to RET. Spoiler shall retract in less than 1.0 second.
(24) Reset AC SPOILERON circuit breaker.

SPOILER AND SPOILER ACTUATOR - MAINTENANCE PRACTICES

1. Removal/Installation

A. Removal of the Spoiler (See Figure 201.)

(1) Connect an external electrical power source to the aircraft.
(2) Connect an external hydraulic power source to the aircraft.
(3) Set the Battery Switches on.
(4) Set the Inverter Switches on.
(5) Set Flap Switch to DN.
(6) Disconnect the external hydraulic power source from the aircraft.
(7) Depressurize the hydraulic system. (Refer to 29-00-00.)
(8) Set the Inverter Switches off.
(9) Set the Battery Switches off.
(10) Remove the attaching parts that attach the actuator arm to spoiler control bracket.
(11) *On Aircraft 35-002 thru 35-240 and 36-002 thru 36-044,* disconnect the bonding jumper from the inboard hinge support.
(12) *On Aircraft 35-241 and Subsequent and 36-045 and Subsequent,* disconnect the bonding jumper from the inboard and the outboard hinge support.
(13) Remove the attaching parts that attach the spoiler hinge to the hinge supports.
(14) Remove the spoiler from the wing.

B. Installation of the Spoiler (See Figure 201.)

NOTE: When a new spoiler assembly is to be installed, refer to the instructions provided in SSK 936 (latest revision).

(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Epoxy Primer		Refer to 20-55-00.	Prime attaching bolts.

(2) Check the condition of the protective tape of the protective coating on the lower edge of the spoiler. Repair or replace the tape or the coating as required. (Refer to 27-60-01.)
(3) Before any attaching parts are installed, inspect all parts (bolts, bearings, bushings, hinges, and nuts, and their riding surfaces for wear, cracks, corrosion, or distortion. Replace any parts that are defective.
(4) Set the spoiler at its correct location on wing.
(5) Install the attaching parts that attach the inboard and outboard spoiler hinges to the hinge supports on the wing.
(6) Set the actuator arm on the spoiler control bracket.
(7) Apply epoxy primer to the spoiler attach bolt before they are installed.
(8) Install the attaching parts that attach the actuator arm to the spoiler control bracket.

(9) Install the attaching parts that attach the bonding jumper(s) to the hinge support(s).
(10) Do the Operational Check of Spoilers. (Refer to 27-60-00.)

C. Removal of the Spoiler Actuator (See Figure 201.)
(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Caps and Plugs (Plastic)		Commercially Available	Cover exposed fittings on hydraulic hoses and spoiler actuator.
Shop Rags		Commercially Available	Catch hydraulic fluid.

(2) Connect an external electrical power source to the aircraft.
(3) Connect an external hydraulic power source to the aircraft.
(4) Set the Battery Switches on.
(5) Set the Inverter Switches on.
(6) Set Flap Switch to DN.
(7) Disconnect the external hydraulic power source from the aircraft.
(8) Depressurize the hydraulic system. (Refer to 29-00-00.)
(9) Set the Inverter Switches off.
(10) Set the Battery Switches off.
(11) Remove spoiler actuator access covers located on aft lower side of wing.

CAUTION: PUT SHOP RAGS UNDER THE HYDRAULIC FITTING OF THE SPOILER ACTUATOR WHEN THE HYDRAULIC HOSES ARE DISCONNECTED.

(12) Put shop rags under the hydraulic fittings on the spoiler actuator to catch any fluid that drips out when the hydraulic hoses are disconnected.
(13) Disconnect the extend and retract hydraulic lines from the spoiler actuator.
(14) Install caps and plugs on the open fittings and hydraulic hoses.
(15) Remove attaching parts that attach the actuator rod end to the actuator arm.
(16) Remove the attaching parts that attach the forward end of the actuator to the attach bracket. Discard the old attaching parts.
(17) Remove the spoiler actuator from the wing.

D. Installation of the Spoiler Actuator (See Figure 201.)
(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Dry Ice or a Refrigerator		Commercially Available	Cold soak the bushing.

CAUTION: EACH TIME THE SPOILER ACTUATOR IS REMOVED OR INSTALLED, NEW HARDWARE MUST BE USED TO ATTACH THE SPOILER ACTUATOR HEAD END TO THE FORWARD ATTACH FITTING.

(2) Replace the bushing in the forward attach bracket as follows:

NOTE: Both of the bushings in the forward attach fitting are press fit.

(a) Remove the old bushings from the forward attach bracket.
(b) Apply epoxy primer to the outside diameter of the flanged bushing.
(c) Install the flanged bushing in the inboard side of the LH forward attach bracket.

NOTE: The flanged bushing is located on the outboard side of the RH froward attach bracket.

(d) Cold soak the non-flanged bushing for approximately 30 minutes.
(e) Remove the bushing from the dry ice and apply epoxy primer to the outside diameter of the bushing.
(f) Install the bushing in the outboard side of the LH forward attach bracket. Make sure that the bushing is centered in the attach bracket flange.

NOTE: The bushing is located on the inboard side of the RH froward attach bracket.

(g) Check and make sure that the spoiler actuator head end bearing will slip between the bushings.

(3) Set the spoiler actuator at its correct location on the forward attach fitting.
(4) Using new attaching parts, attach the spoiler to the forward attach fitting.
(5) *On Aircraft 35-002 thru 35-427 and 36-002 thru 36-049,* torque the nut 65 to 90 in lb [7.33 to 10.16 Nm] (including drag torque).
(6) *On Aircraft 35-428 and Subsequent, 36-050 and Subsequent,* torque the nut to 30 in lb [3.36 Nm] (minimum) above run-in torque. Tighten to next castellation and install cotter pin.
(7) Set the opposite end of the spoiler actuator at its correct location on the actuator arm.
(8) Install the attaching parts that attach the spoiler actuator to the actuator arm.
(9) Remove the caps and plugs from the hydraulic hoses and fittings.
(10) Install the hydraulic hoses on the spoiler actuator.
(11) Remove the shop rags from the wing.
(12) Bleed spoiler system to remove any air trapped in the system.
(13) Rig the spoilers. (Refer to 27-60-00.)

2. Repairs

A. Replacement of the Protective Tape or Sealant

NOTE: Some aircraft may have tape installed along the edge of the lower surface of the spoiler to protect the finish on the flap.

Some aircraft may have a protective coating of sealant installed along the edge of the lower surface of the spoiler to protect the finish on the flap.

Service history has shown that the tape may peel off during flight. If the tape is peeling, it is permissible to use a coat of sealant to replace the tape.

(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Protective Tape	#8681	3M Company St. Paul, MN	Prevent spoiler/flap chafe condition.
Isopropyl Alcohol		Commercially Available	Clean surface of spoiler.
Adhesion Promoter	#86	3M Company St. Paul, MN	Prepare surface for tape.
Lint-Free Cloth		Commercially Available	Clean surface of spoiler.
Sealant	Pro-Seal 890	Refer to 20-10-00.	Prevent spoiler/flap chafe condition.
Lubricant	LPS #1	LPS Research Lab. Inc. Tucker, GA	Lubricant for protective coating.
Rubber Roller		Commercially Available	Smooth out the tape.
Abrasive Pad.		Commercially Available	Clean surface of the spoiler.
Chemical Film Agent	Refer to 20-72-00.		Protect the surface of the spoiler.
Cleaning Solvent	Refer to 20-13-00.		Clean surface of the spoiler.

(2) If protective tape is damaged or peeling, replace as follows:
   (a) Clean any oxides from the lower spoiler surface from trailing edge forward 1.5 inches [3.80 cm] with fine abrasive pad.
   (b) Touch up the chemical film. (Refer to 20-72-00.)

(c) Clean surface with a 50/50 mixture of water and isopropyl alcohol.
(d) After surface is dry, apply a light coat of adhesion promoter with a clean, lint-free cloth.
(e) Allow promoter to dry 15 to 20 minutes.
(f) Install the tape to the lower spoiler surface aligning edge of tape with trailing edge of spoiler.
(g) Use a roller to work out any bubbles being careful not to stretch tape.

(3) If sealant coating is damaged, replace as follows:
(a) Remove the damaged area or entire strip of sealant coating.
(b) Solvent clean lower surface of spoiler surface. (Refer to 20-12-00.)
(c) Apply a thin coat of sealant 0.80 [± 0.20] inch [2.03 (± 0.51) cm] wide to lower edge of spoiler.
(d) Allow sealant to cure.
(e) Apply light coat of lubricant to strip of sealant.

B. Replacement of the Spoiler Actuator Arm
(1) Remove the spoiler assembly from the wing. (Refer to 27-60-01.)
(2) Remove the attaching parts that attach the actuator arm to the wing.
(3) Remove the attaching parts that attach the actuator arm to the spoiler actuator.
(4) Remove the bearing from the actuator arm. Replace the bearing if necessary.
(5) Using the pilot holes in the actuator arm, step ream the holes to 0.312 inch [7.924 mm].
(6) Install the attaching parts that attach the actuator arm to the spoiler actuator.
(7) Install the bearing in the actuator arm.
(8) Install the attaching parts that attach the actuator arm to the wing.
(9) Install the spoiler assembly. (Refer to 27-60-01.)

C. Disassemble of the Spoiler Actuator (See Figure 202.)
(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Stoddard Solvent		Commercially Available	Clean actuator parts.
Crocus Cloth		Commercially Available	Polish actuator parts.
Oil Stone		Commercially Available	Remove nicks and scratches.

(2) Remove the safety wire from the rod end.
(3) Loosen the jamnut sufficiently to release rod end lock.
(4) Remove rod end from piston rod.
(5) Mark the clocked position of head end on barrel assembly and number of threads protruding from barrel assembly before removing head end.
(6) Loosen the locknut on the head end sufficiently to allow head end to be screwed out of barrel assembly.

CAUTION: AS THE PISTON IS REMOVED FROM THE BARREL ASSEMBLY, THE LOCKING BALLS WILL BE RELEASED. TAKE PRECAUTIONS TO CATCH THEM.

(7) Slowly remove the piston from the barrel assembly. Make sure that you catch the locking balls.
(8) Remove O-rings, backup rings, and, if applicable, shim.

WARNING: THE PLUNGER IS SPRING-LOADED IN THE HEAD END. EXTREME CARE MUST BE TAKEN WHEN REMOVING PLUNGER FROM THE HEAD END TO PREVENT EITHER PART FROM SPRINGING OUT OF CONTROL. OTHERWISE BODILY INJURY MAY RESULT.

(9) Remove retaining ring, plunger, and spring from head end.
(10) Remove O-rings and backup rings from plunger.
(11) Remove felt wiper, O-ring, and backup ring from barrel assembly.
(12) Clean all parts with Stoddard solvent, or equivalent.
(13) Dry parts with clean, dry, compressed air.
(14) Inspect the barrel assembly, piston rod, plunger, and head end for nicks, scratches, or excessive wear.
(15) Buff out minor defects with an oil stone. Polish with crocus cloth.

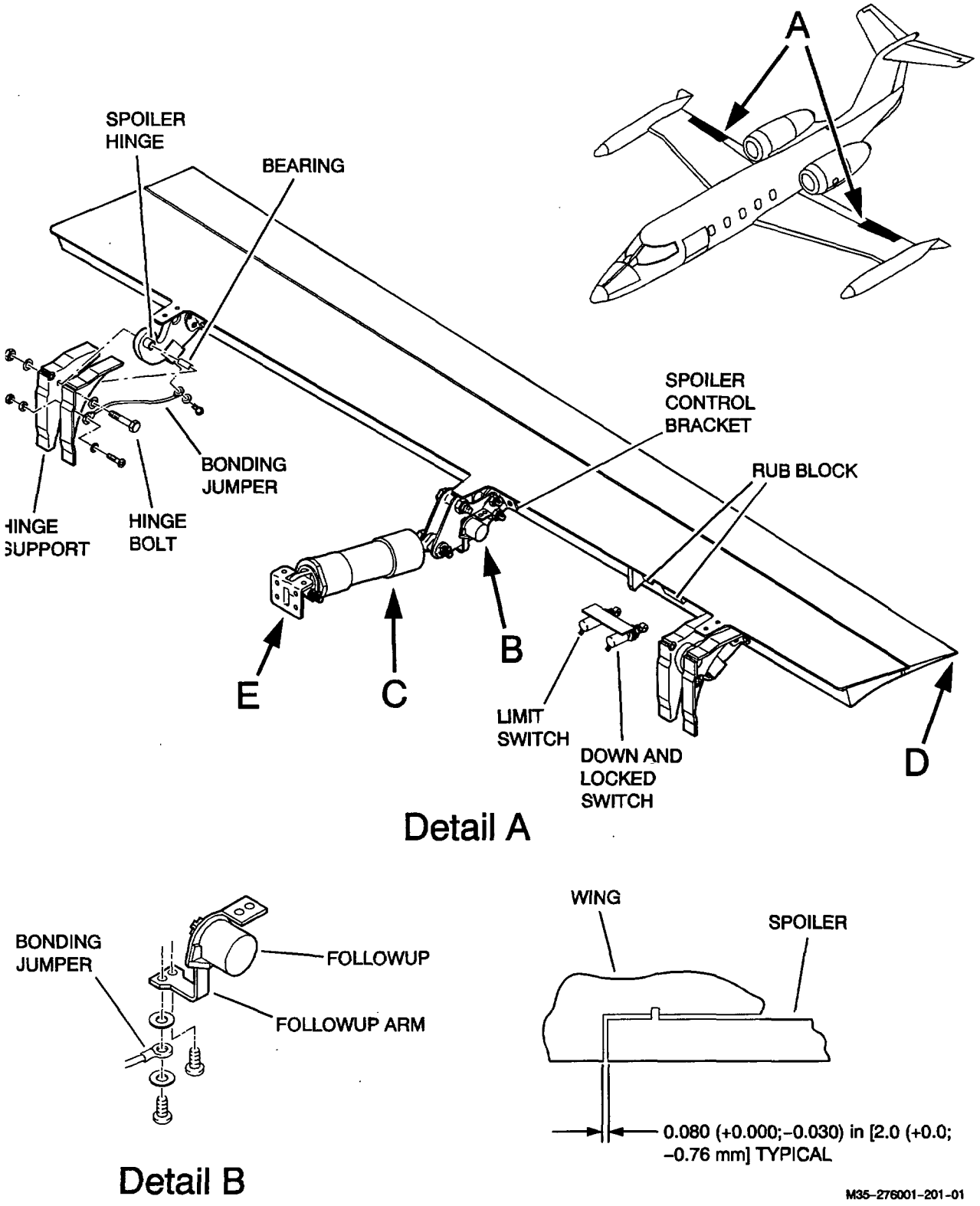
D. Assembly of the Spoiler Actuator (See Figure 202.)
(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Safety Wire	MS20995F32	Commercially Available	Safety wire the jamnut and rod end.
Petroleum Jelly (Vaseline)	AN-P-51	Commercially Available	Hold the locking balls in place during assembly.
Hydraulic Fluid	MIL-H-5606	Commercially Available	Soak assembly parts.
Hydraulic Test Bench (Capable of 2250 psi [1551 kPa] maximum)		Commercially Available	Leak check of the spoiler actuator.

(2) Soak the new O-rings, backup rings, and felt wiper in hydraulic fluid immediately before installation.
(3) Install new backup ring, new O-ring, and new felt wiper in barrel assembly.
(4) Install new O-rings and new backup rings on plunger. Insert spring into plunger cavity.
(5) Insert plunger and spring into head end and attach with retainer ring.
(6) Install new O-rings and new backup rings on piston rod.
(7) Install shim, if applicable, on piston rod. Ensure that piston rod stroke is 1.625 to 1.645 in [41.275 to 41.783 mm]. Add or subtract the shims as required to maintain this dimension.

(8) Coat the outside and the inside diameters of piston rod with petroleum jelly to hold the locking balls in the holes in the piston rod.
(9) Install locking balls in piston rod.
(10) Carefully insert piston rod into barrel assembly. Make sure that the locking balls are not dislodged.
(11) Screw the head end into barrel assembly. Make sure that you leave the same number of threads exposed and so that clevis end is clocked as it was before.
(12) Install the locknut.
(13) Install rod end, rod end lock, and jamnut on piston rod.
(14) Install the safety wire on the rod end lock.
(15) Connect the actuator to a hydraulic test bench and cycle several times at 2250 psi [15511 kPa] (maximum).
(16) The maximum allowable leakage is one (1) drop.

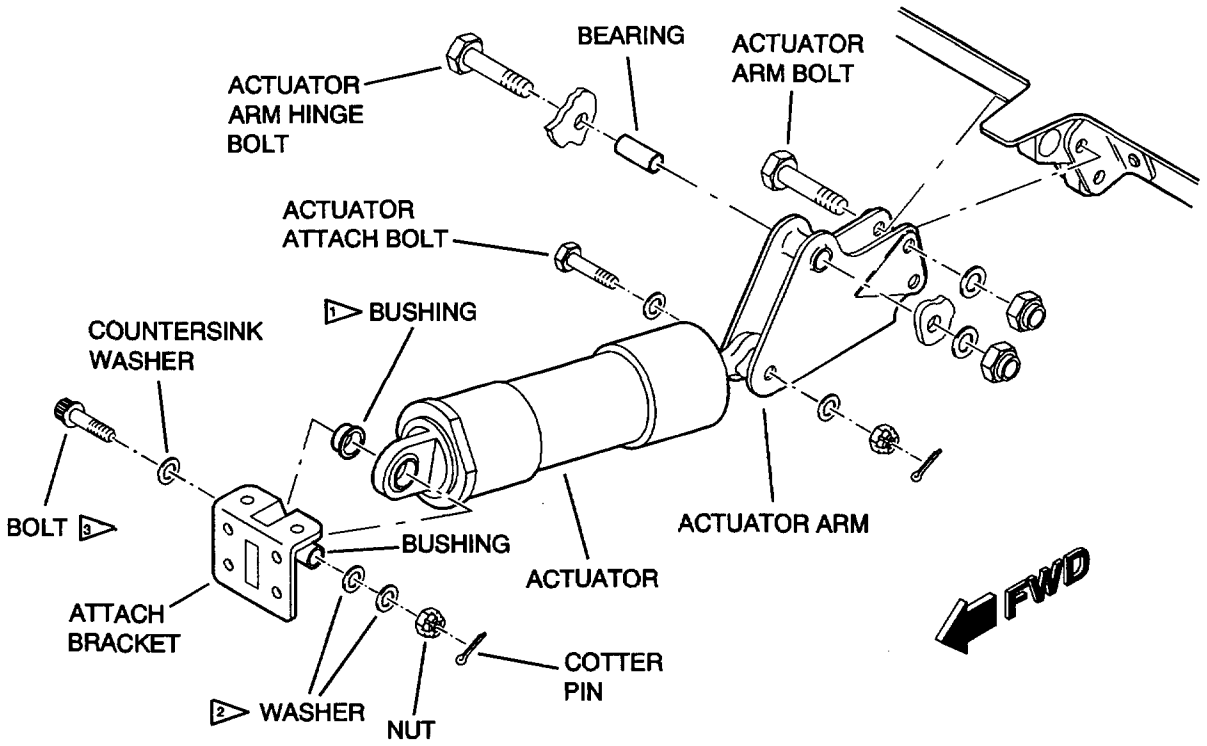


Spoiler and Spoiler Actuator Installation
Figure 201 (Sheet 1 of 3)

1 INSTALL BUSHINGS WITH WET EPOXY PRIMER.

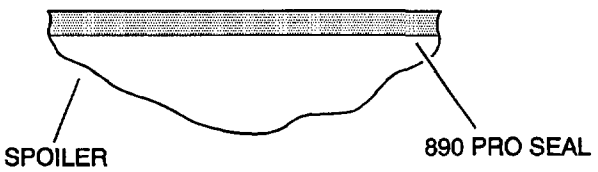
2 DEPNDING UPON BOLT LENGTH, INSTALL WASHERS AS REQUIRED FOR NUT TO HAVE ADEQUATE THREAD ENGAGEMENT AND TO BE PROPERLY TORQUED.

3 INSTALL BOLT HEAD INBOARD.



(TYPICAL FOR BOTH WINGS)

Detail C



Detail D

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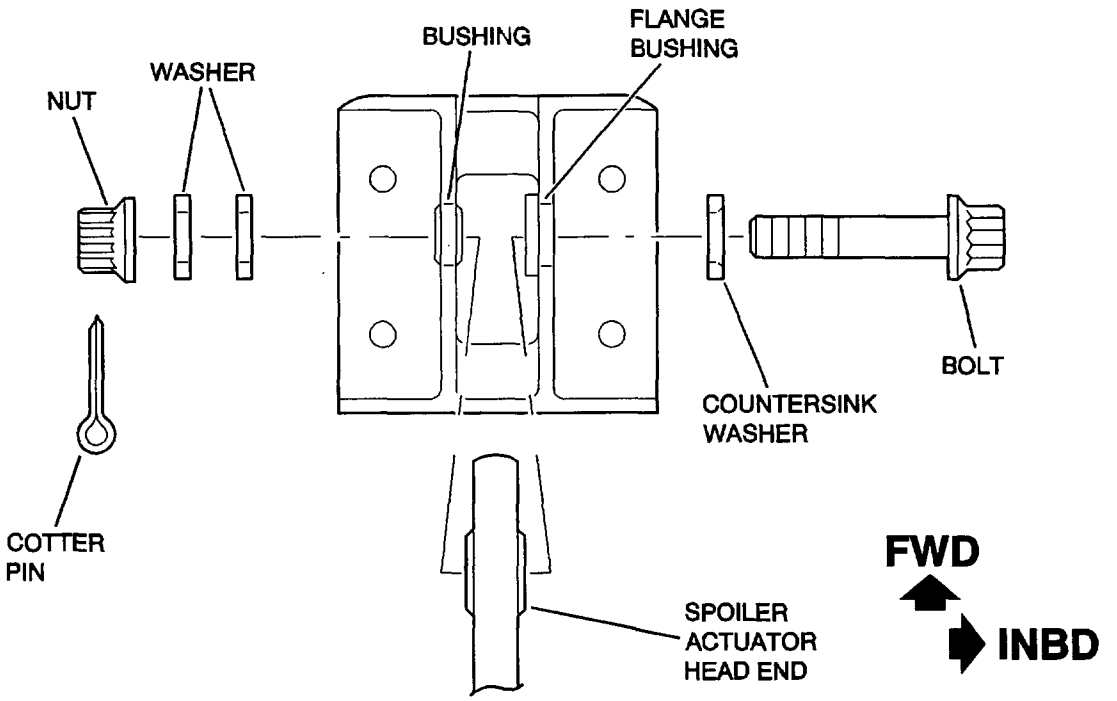
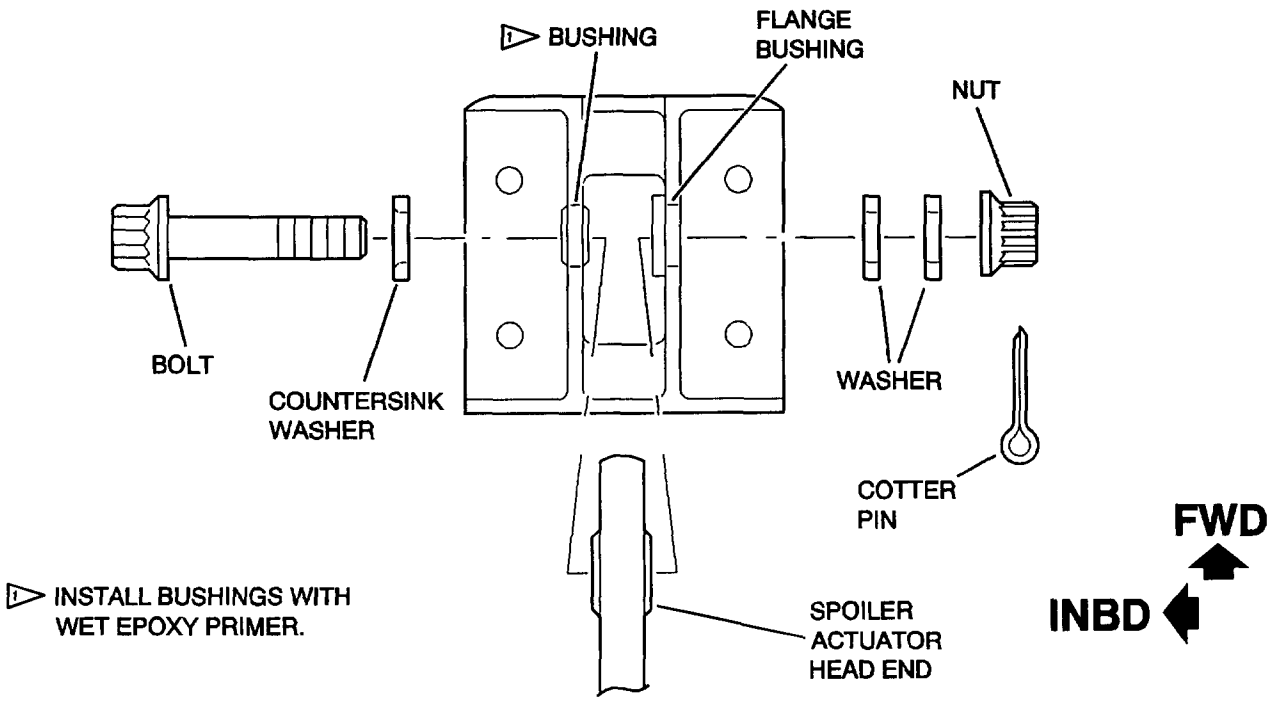
Spoiler and Spoiler Actuator Installation
Figure 201 (Sheet 2 of 3)



VIEW LOOKING FORWARD AT LH WING



VIEW LOOKING FORWARD AT RH WING

Detail E

Spoiler and Spoiler Actuator Installation

Figure 201 (Sheet 3 of 3)

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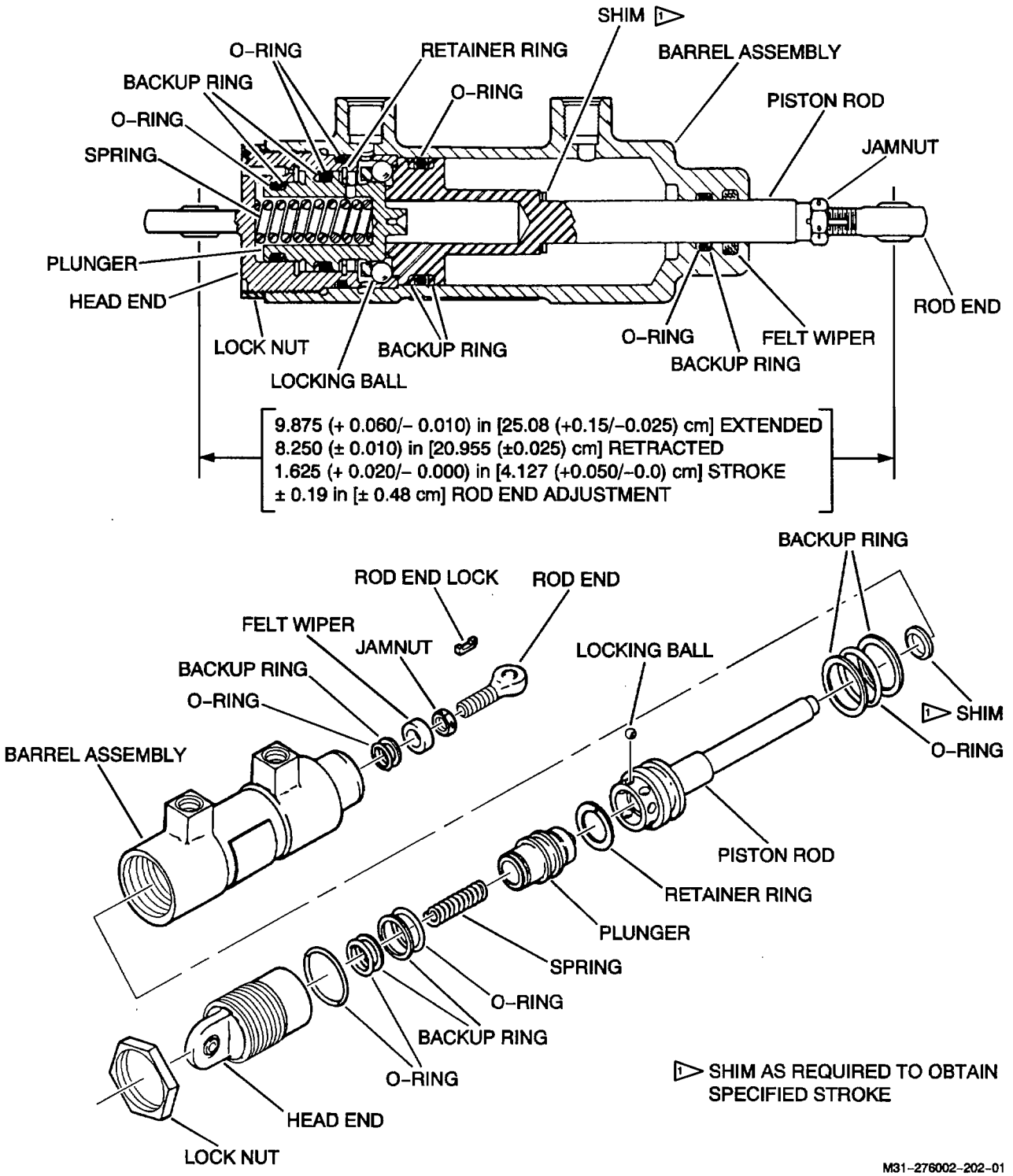


Spoiler Actuator Assembly
Figure 202

SPOILER SYSTEM FILTER - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

NOTE: ° The spoiler system filter element must be removed and cleaned or replaced in accordance with current inspection intervals. (Refer to Chapter 5.)

° The filter is located on the centerline of the aircraft and is accessible through the right landing gear wheel well.

° Removal of the entire filter assembly is not necessary when cleaning or replacing the filter element. Only the filter bowl and element need be removed.

A. Remove Filter Element (See figure 201.)

WARNING: FAILURE TO DISCONNECT AIRCRAFT BATTERIES COULD RESULT IN BODILY INJURY WHILE WORKING IN THE WHEEL WELL AREA IF ELECTRICAL POWER IS INADVERTENTLY APPLIED TO THE LANDING GEAR.

(1) Lower tailcone access door and disconnect aircraft batteries.
(2) Depressurize hydraulic system. (Refer to Chapter 29.)
(3) Using a screwdriver to override the uplatch actuator, release uplatch hook from roller at right inboard gear door.
(4) To lower the main gear inboard door, station a man at the inboard gear door and a man in the cockpit to operate the brakes. With one man pumping the brakes, the man stationed at the door must pull the door to the down position.

NOTE: ° The brakes must be pumped during manual lowering of the inboard gear door to prevent possible door actuator damage and/or nose gear vacuuming to up cycle.

° On Aircraft equipped with cleanable type element, proceed with steps A.(5) thru A.(9). On Aircraft equipped with throwaway type element, proceed with steps A.(10) thru A.(13).

(5) Cut and remove safety wire from filter bowl and filter head.
(6) Using a one-inch wrench, unscrew and remove filter bowl.
(7) Pull straight down on filter element to remove it from filter head.
(8) Remove and discard O-ring located inside of element neck.
(9) Remove and discard O-ring and backup ring located at the base of the bowl threads.
(10) Cut and remove safety wire from filter bowl and filter head.
(11) Unscrew and remove filter bowl.
(12) Unscrew filter element from filter head. Remove filter element O-ring. Discard both filter element and O-ring.

(13) Remove filter bowl O-ring and backup rings. Check condition of O-ring and backup rings and replace if necessary.

B. Clean Filter Element

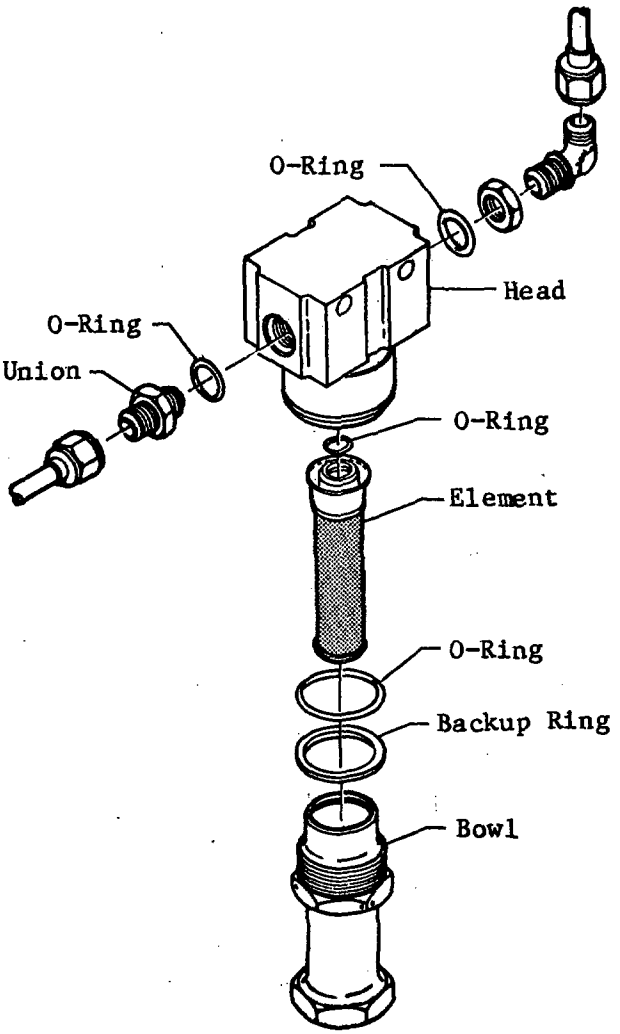
(1) Fill filter element cavity with full strength liquid detergent (biodegradable).
(2) Insert a rubber plug with a hole through the center into the element neck and, using clean dry air, force the detergent out through the element.
(3) Remove the rubber plug and reverse flush element with tap water to remove soap bubbles.
(4) Reinsert the rubber plug and blow water out of element.
(5) Remove rubber plug and reverse flush element with tap water until water runs clear from element.
(6) Again fill element cavity with full strength liquid detergent and repeat steps (2) through (5) until element is clean.
(7) Dry element with clean dry air or nitrogen.

C. Install Cleanable Type Filter Element

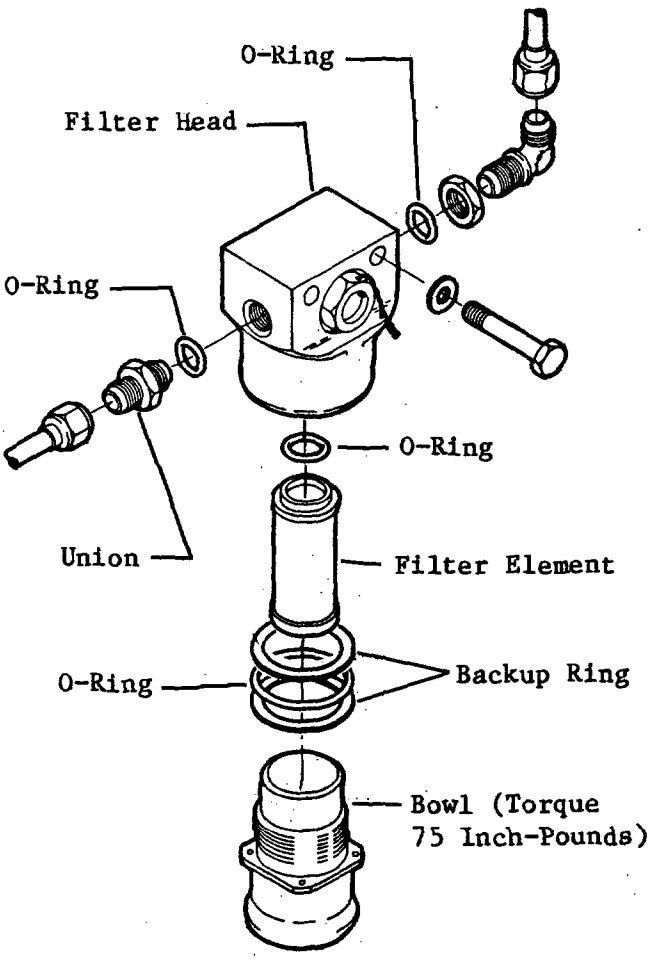
(1) Lubricate new O-rings and backup ring with MIL-H-5606 hydraulic fluid and install.
(2) Position filter element in bowl and screw bowl into head.
(3) Torque filter bowl 50 to 60 inch-pounds and safety wire per double twist method.
(4) Reconnect aircraft batteries and secure tailcone access door.

D. Install Throwaway Type Filter Element

(1) Lubricate new O-rings and backup rings with MIL-H-5606 hydraulic fluid. Install O-ring and backup rings in filter head.
(2) Install filter element O-ring and filter element. Tighten element firmly.
(3) Install filter bowl and torque to 75 inch-pounds. Safety wire.



Filter Assembly - Cleanable Element



Filter Assembly - Throwaway Element

Spoiler System Filter Assembly
Figure 201

SPOILER SERVO VALVE MANIFOLD - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
28 VDC Source		Commercially Available	Electrical supply.
Multimeter	260-7	Simpson	Resistance and voltage checks.
Hydraulic Power Source (1500 psig, 2.0 gpm Minimum)		Commercially Available	Hydraulic supply.
Pressure Gage (3000 psig [20684 kPa])		Commercially Available	Measure pressure.
10 µ Absolute Filter (2 gpm [7.57 lpm] Minimum Rating)		Commercially Available	Filter hydraulic fluid.
Needle Valve		Commercially Available	Test manifold.
Solenoid Operated Selector Valve		Commercially Available	Test manifold.
Test Manifold*	MIT62137	Learjet Corp. Wichita, KS	Test manifold.
Test Blocks*	MIT62136	Learjet Corp. Wichita, KS	Test manifold.
Stoddard Solvent	PS-661	Commercially Available	Cleaning.
Hydraulic Fluid	MIL-M-5606	Commercially Available	Lubrication.
Safety Wire		Commercially Available	Safetying parts.
Torque Wrench (0 to 200 in-lb[0-25 kPa])		Commercially Available	Torquing.

*Only *Aircraft 35-117 and Subsequent, 36-033 and Subsequent*.

2. **Approved Repairs** (*35-002 thru 35-116, 36-002 thru 36-032*)

NOTE: The spoiler servo valve manifold assembly is bolted to the forward side of wing spar 8 immediately to the right of the keel beam.

Access to the manifold attach bolts is gained through the second access cover aft of the right inboard gear door.

Access to the manifold assembly and associated plumbing is gained through the inboard most access cover on spar 7 in the right main gear wheel well.

All hydraulic fittings must be capped immediately after being disconnected to avoid contamination.

The internal cleanliness of the manifold assembly is critical due to the sensitivity of the servo valves to contamination. When performing maintenance on the manifold assembly, it is imperative that all components be kept scrupulously clean.

No lubricants are permitted except MIL-H-5606 hydraulic fluid.

The only maintenance to the servo valve that is allowed is replacement. Under no circumstances should the servo valve be opened up. Servo valves requiring repair must be returned to Learjet Spares Department.

A manifold assembly repair kit is available and contains replacement O-rings and backup rings.

A. Removal of Manifold Assembly (See Figure 201.)

WARNING: FAILURE TO DISCONNECT AIRCRAFT BATTERIES COULD RESULT IN BODILY INJURY WHILE WORKING IN THE WHEEL WELL AREA IF ELECTRICAL POWER IS INADVERTENTLY APPLIED TO THE LANDING GEAR SYSTEM.

(1) Lower tailcone access door and disconnect aircraft batteries.
(2) Using a screwdriver to override uplatch actuator, release uplatch hook from roller at right main gear inboard door.
(3) To lower the main gear inboard door, station a man at the door and a man in the cockpit to operate the brakes. With one man pumping the brakes, the man stationed at the gear door must pull the door to the down position.

WARNING: THE BRAKES MUST BE PUMPED DURING MANUAL LOWERING OF THE INBOARD GEAR DOOR TO PREVENT POSSIBLE DOOR ACTUATOR DAMAGE AND/OR NOSE GEAR VACUUMING TO UP CYCLE.

(4) Depressurize aircraft hydraulic system. (Refer to 29-00-00.)
(5) Remove inboard-most access cover on spar 7 in the right main gear wheel well.
(6) Disconnect wiring from servo valves by cutting wires at splice.
(7) Disconnect and cap plumbing from manifold assembly.
(8) Remove the second access cover aft of the right inboard gear door to gain access to manifold attach bolts on the aft side of spar 8.
(9) Remove manifold attach bolts and remove the manifold assembly through spar 7 access.

B. Disassemble Manifold Assembly (See Figure 201.)
(1) Remove the four screws attaching each servo valve to manifold. Remove and discard the O-rings from the base of servo valve. Place servo valves in separate clean plastic bag.

(2) Remove unions from ports A and B. Remove and discard O-rings from unions.
(3) Cut and remove safety wire from plugs.
(4) Unscrew and remove plugs from manifold. Remove blind sealed sleeve, check valve and plain sleeve from each port where plugs were removed. Remove and discard backup rings and O-rings from blind sealed sleeve and check valve.
(5) Unscrew and remove reducers from manifold. Remove and discard O-ring.
(6) Unscrew and remove threaded sleeve from each port. Remove check valve and plain sleeve from each port. Remove and discard O-rings and backup ring from check valve.

NOTE: It will help if the parts from each port are put into separate containers and labeled with their respective port identification. When removing check valves, sketches showing port identification and flow direction will be helpful in reassembly.

C. Assemble Manifold Assembly (See Figure 201.)
(1) Thoroughly clean all parts in Stoddard solvent and dry with clean dry compressed air.

NOTE: The internal cleanliness of the manifold assembly is critical due to the sensitivity of the servo valves to contamination. It is imperative that all components be kept scrupulously clean.

(2) Immerse all O-rings and backup rings in MIL-H-5606 hydraulic fluid prior to installation.
(3) Install new O-rings and backup rings on check valves.
(4) Install plain sleeve in port marked 1LU. Install check valve into same port with flow direction toward the plain sleeve. Screw threaded sleeve into port approximately 1/2 inch (1.27 centimeters).
(5) Install check valve into port marked 1RU with flow direction toward the plain sleeve. Screw threaded sleeve into port approximately 1/2 inch (1.27 centimeters).
(6) Alternately screw in threaded sleeves until components bottom out. Using a depth gage, ensure that both threaded sleeves are the same depth. This will ensure that components are centered. Torque threaded sleeves 200 (±10) inch-pounds (22.58 ± 1.13 Nm).
(7) Install new O-rings on reducers and install into ports marked 1LU and 1RU.
(8) Install plain sleeve into port marked 1LD. Install check valve into same port with flow direction away from the plain sleeve. Screw threaded sleeve into port approximately 1/2 inch (1.27 centimeters).
(9) Install check valve into port marked 2RD with flow direction away from plain sleeve. Screw threaded sleeve into port approximately 1/2 inch (1.27 centimeters).
(10) Alternately screw in threaded sleeves until components bottom out. Using a depth gage, ensure that both threaded sleeves are the same depth. This will ensure that the components are centered. Torque threaded sleeves 200 (±10) inch-pounds (22.58 ± 1.13 Nm).
(11) Install new O-rings on reducers and install into ports marked 2LD and 2RD.
(12) Install new O-rings on unions and install unions in ports A and B.
(13) Install new O-rings and backup rings on blind sealed sleeves and their respective check valves.
(14) Install plain sleeve into hole next to port labeled 1LU. Install check valve into the same hole with flow direction directed away from plain sleeve. Install blind sealed sleeve and screw plug partially into manifold.
(15) Install check valve into hole next to port labeled 1RU. Ensure that flow direction of check valve is away from plain sleeve installed in step (14). Install blind sealed sleeve and screw plug partially into manifold.
(16) Alternately screw plugs in until components bottom against each other. Ensure that both plugs extend equally from end of manifold; this centers the column of components within the manifold.
(17) Torque plugs 200 (±10) inch-pounds (22.58 ± 1.13 Nm).
(18) Install plain sleeve into hole next to port labeled 2LD. Install check valve into same hole with flow direction directed toward the plain sleeve. Install blind sealed sleeve and screw plug partially into manifold.

(19) Install check valve into hole next to port labeled 2RD. Ensure that flow direction of check valve is toward plain sleeve installed in step (18). Install blind sealed sleeve and screw plug partially into manifold.
(20) Alternately screw plugs in until components bottom against each other. Ensure that both plugs extend equally from end of manifold; this centers the column of components within the manifold.
(21) Torque plugs 200 (±10) inch-pounds (22.58 ±1.13 Nm).
(22) Safety wire plugs using double twist method.
(23) Functional test manifold. (Refer to Adjustment/Test.)
(24) Remove plugs from servo valve and position new O-rings into ports at bottom of servo valve.
(25) Position servo valve on manifold so that index pin on servo valve aligns with index hole in manifold. Torque screws evenly to ensure even O-ring compression. Repeat for remaining servo valve.
(26) Functional test servo valve manifold assembly. (Refer to Adjustment/Test.)

D. Installation of Servo Manifold Assembly (See Figure 201.)
(1) Position manifold assembly on spar 8, ensuring that port A is up and align bolt holes. Secure manifold assembly with bolts.
(2) Remove plugs and connect plumbing to manifold. Connect wiring.
(3) Replace access covers.
(4) Using an auxiliary power unit, cycle spoiler system a minimum of five times to expel air from system. Do not use auxiliary hydraulic pump for this purpose.
(5) Connect aircraft batteries and secure tailcone access door.
(6) Perform operational check of spoiler system. (Refer to 27-60-00, Inspection/Check.)

3. **Approved Repairs** (*35-117 and Subsequent, 36-033 and Subsequent*)

NOTE: The spoiler servo valve manifold assembly is bolted to the forward side of wing spar 8 immediately to the right of the keel beam.

Access to the manifold attach bolts is gained through the second access cover aft of the right inboard gear door.

Access to the manifold assembly and associated plumbing is gained through the inboard most access cover on spar 7 in the right main gear wheel well.

All hydraulic fittings must be capped immediately after being disconnected to avoid contamination.

The internal cleanliness of the manifold assembly is critical due to the sensitivity of the servo valves to contamination. When performing maintenance on the manifold assembly, it is imperative that all component be kept scrupulously clean.

No lubricants are permitted except MIL-H-5606 hydraulic fluid.

The only maintenance permitted on the servo valves is replacement.

The maintenance procedures for the servo valve manifold include only removal, installation and functional test of the servo valves. All other maintenance performed on the manifold shall be in accordance with the vendor's overhaul instructions.

A. Removal of Servo Valves from Manifold Assembly (See Figure 202.)
(1) Remove servo valves electrical connector from clamp.

(2) Remove the four screws attaching each servo valve to manifold. Remove and discard the O-rings from the base of servo valve.
(3) Plug ports in manifold and ports in servo valves.
(4) Place each servo valve in a separate, clean plastic bag.

B. Installation of Servo Valves on Manifold Assembly (See Figure 202.)
(1) Remove plugs from servo valve and position new O-rings into ports at bottom of servo valve.
(2) Remove plugs from manifold and position servo valves on manifold so that index pin on servo valve aligns with index hole in manifold.
(3) Install four screws securing servo valve to manifold. Torque evenly to ensure even O-ring compression.
(4) Replace electrical connector in clamp and secure.

C. Perform Functional Test of Servo Valves as follows:
(1) Assemble servo valve on test manifold using new O-rings.
(2) Plug all ports in the manifold except ports P and R.
(3) Apply 1500 psig (10342 kPa) to port P. Leakage from port R shall not exceed 0.25 gpm (0.95 lpm).
(4) Remove pressure from port P. Remove plugs from all ports.
(5) Apply 1500 psig (10342 kPa) to port P. Flow is permissible from ports C2 and R. No leakage is allowed from port C1 after 2 minute wait.
(6) Apply +20 ma max. D.C. power to the red wire on the servo valve. C2 flow will increase to full bench flow.
(7) Removal of power from red wire and apply power to green wire. C2 flow shall stop and C1 flow will go to full bench flow and should equal C2 flow in step (6).
(8) Remove pressure and power from servo valve and disassemble servo valve from test manifold.
(9) Repeat steps (1) through (8) for the remaining servo valve.

4. Adjustment/Test (*35-002 thru 35-116, 36-002 thru 36-032*)

NOTE: Functional testing of the servo valves and manifold assembly is necessary after disassembly of the manifold assembly.

Functional testing of the servo valves and manifold includes testing the manifold without servo valves, and testing the manifold and servo valves assembled together.

A. Perform functional test of manifold as follows:
(1) Remove servo valves from manifold and plug ports in servo valves.
(2) Install test blocks on manifold using new O-rings.
(3) Using tubing or hose, connect port C1 to port R on each test block.
(4) Apply 2250 psig (15513 kPa) to ports A and B on the manifold with all other ports blocked. There shall be no leakage.
(5) Remove plumbing from test blocks and plugs from manifold ports.
(6) Apply test bench pressure to manifold port B with all other ports open. Flow shall be free and unrestricted from manifold ports 2RD and 2LD and port C2 of each test block.
(7) Remove test bench pressure. Plug ports C1, C2, and P on each test block and all manifold ports except port B. This leaves test block ports R open and manifold port B open.
(8) Apply 1500 psi (10342 kPa) to port B. Leakage from ports R shall not exceed one drop in five minutes after a 2-minute wait.
(9) Remove pressure from port B. Plug test block port R. Remove plug from port 2LD.
(10) Apply 1500 psi (10342 kPa) to port 2LD. Leakage from port B shall not exceed one drop in five minutes after a 2-minute wait.
(11) Remove pressure from port 2LD and plug port 2LD. Remove plug from port 2RD.
(12) Apply 1500 psi (10342 kPa) to port 2RD. Leakage from port B shall not exceed one drop in five minutes after 2-minute wait.

(13) Remove pressure from and plug port 2RD. Remove plug from port R of the left test block.
(14) Apply test bench pressure to port R of the left test block. Flow shall be free and unrestricted from port B.
(15) Remove pressure from and plug port R of the left test block. Remove plug from port R of the right test block.
(16) Apply test bench pressure to port R of the right test block. Flow shall be free and unrestricted from port B.
(17) Remove pressure from port R of right test block. Remove plugs from port A of each test block and from ports 1LU and 1RU on manifold. Plug port B.
(18) Apply 1500 psi (10242 kPa) to port A. Leakage from ports 1LU and 1RU shall not exceed one drop in five minutes after a 2-minute wait.
(19) Remove pressure from port A. Plugs ports 1LU and 1RU. Remove plug from port P of the left test block.
(20) Apply 1500 psi to port P of the left test block. Leakage from port A shall not exceed one drop in five minutes after 2-minute wait.
(21) Remove pressure from and plug port P of the left test block. Remove plug from port P of the right test block.
(22) Apply 1500 psi to (10342 kPa) port P of the right test block. Leakage from port A shall not exceed one drop in five minutes after 2-minute wait.
(23) Remove pressure from and plug port P of the right test block. Remove plug from port 1LU.
(24) Apply test bench pressure to port 1LU. Flow shall be free and unrestricted from port A.
(25) Remove pressure from and plug port 1LU. Remove plug from port 1RU.
(26) Apply test bench pressure to port 1RU. Flow shall be free and unrestricted from port A.
(27) Remove pressure from port 1RU.
(28) Remove test blocks and associated plumbing from manifold. Plug ports of manifold to prevent foreign material from contaminating manifold.

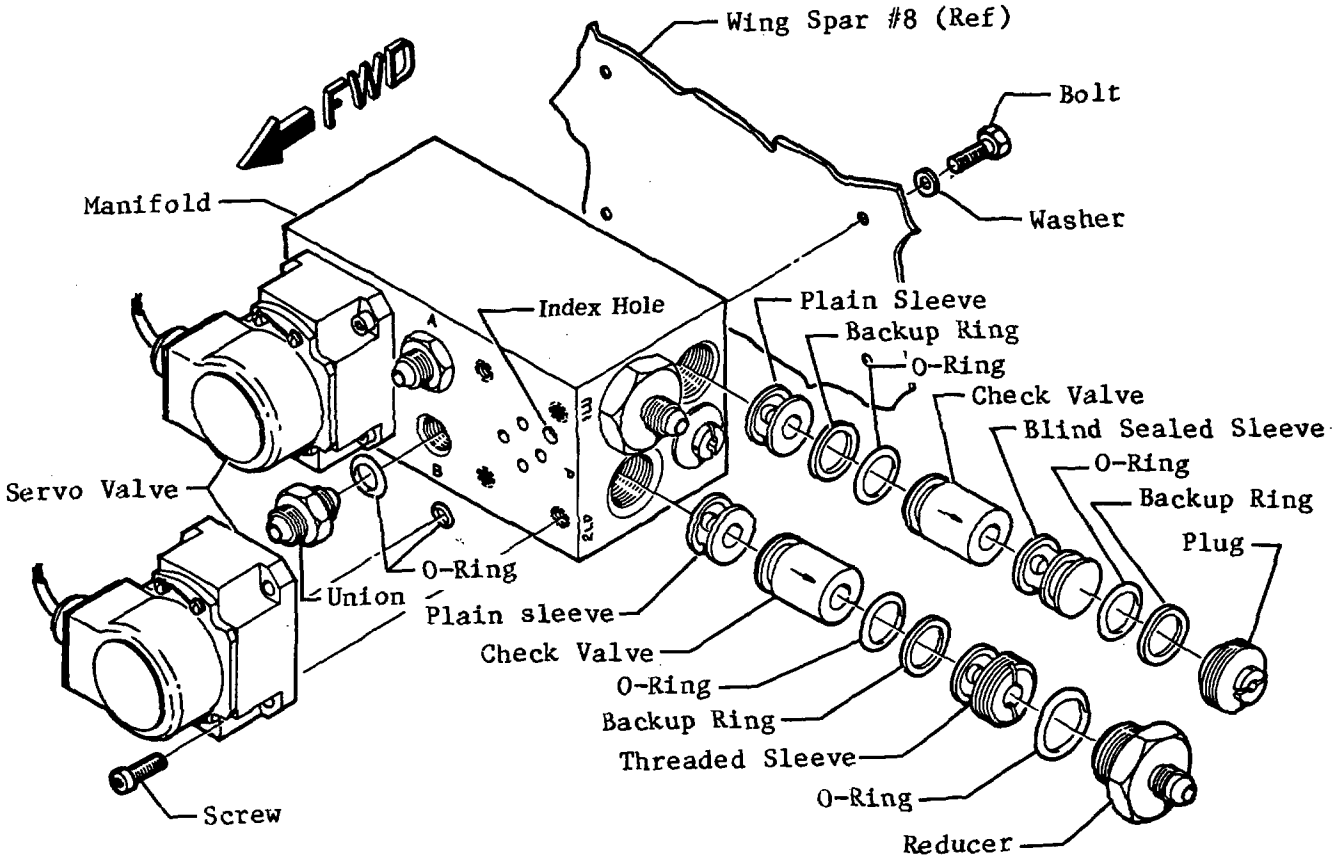
B. Perform Functional Test of Servo Valves as follows:
(1) Assemble servo valve on test manifold using new O-rings.
(2) Plug all ports in the manifold except ports P and R.
(3) Apply 1500 (10342 kPag) psig to port P. Leakage from port R shall not exceed 0.25 gpm (0.95 lpm).
(4) Remove pressure from port P. Remove plugs from all ports.
(5) Apply 1500 psig (10342 kPag) to port P. Flow is permissible from ports C2 and R. No leakage is allowed from port C1 after 2-minute wait.
(6) Apply +20 ma max. DC power to the red wire on the servo valve. C2 flow will increase to full bench flow.
(7) Remove power from red wire and apply power to green wire. C2 flow shall stop and C1 flow will go to full bench flow and should equal C2 flow in step (6).
(8) Apply 1500 psig on port P with all ports open. Apply ± 20 ma max. DC signal to the red wire and listen for smooth flow through the valve.
(9) Apply ± 20 ma max. DC power to the green wire and listen for smooth flow through the valve.

NOTE: Any mechanical chatter sounds indicating a dithering valve position shall be cause for rejection.

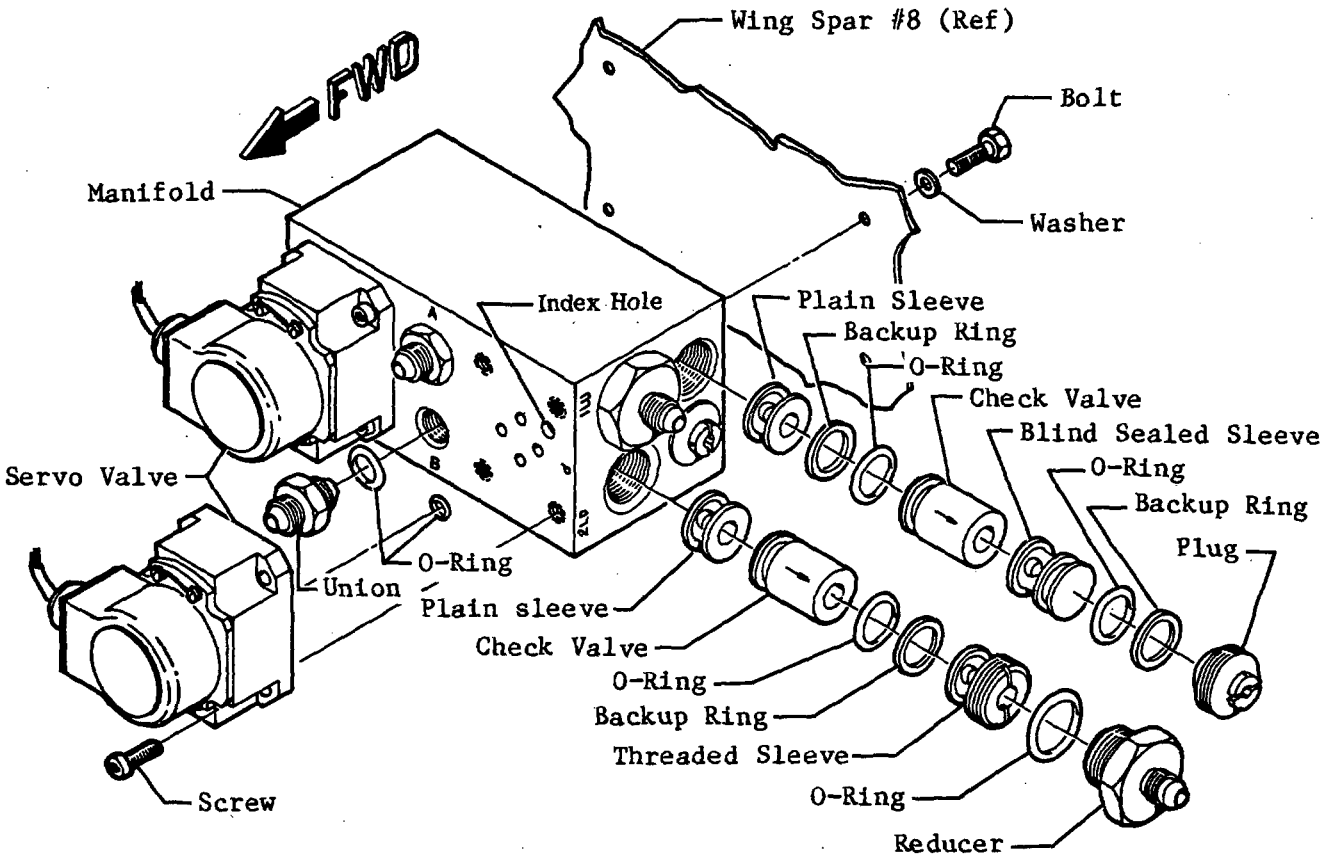
(10) Repeat the test at 1300 psig, 1100 psig, 900 psig, 700 psig, 500 psig.
(11) Remove pressure and power from servo valve and disassemble servo valve from test manifold.
(12) Repeat steps (1) through (8) for the remaining servo valve.

C. Perform Functional Test of Servo Valve and Manifold Assembly as follows:
(1) Install servo valves on manifold using new O-rings.
(2) Plug all ports in the manifold except ports A and B.
(3) Apply 2250 psig (15513 kPa) to both ports A and B. There shall be no leakage after 5 minutes.

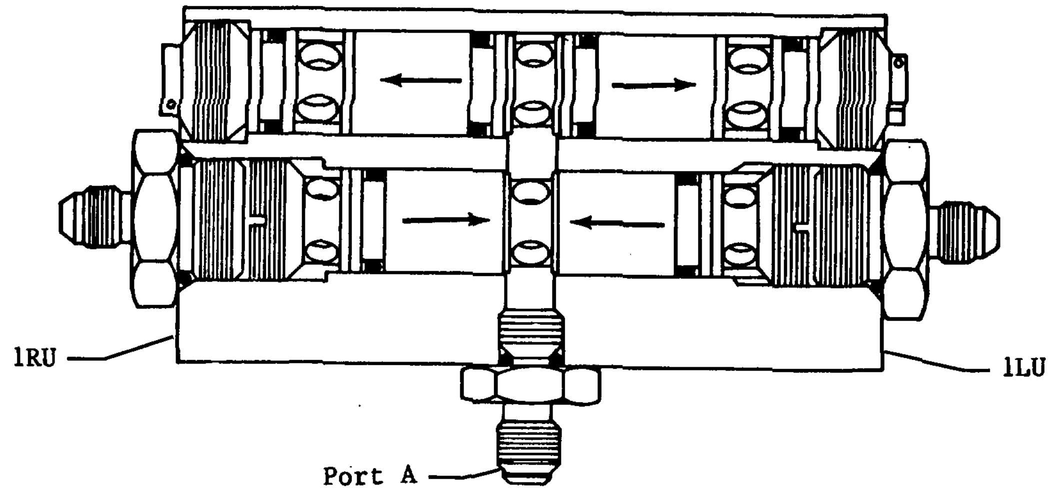
(4) Remove pressure and plugs from ports.
(5) Apply 1500 psig (10342 kPag) to port A with port R plugged. Flow shall result from ports 2LD and 2RD.
(6) Apply +20 ma max. DC power to the red wire on the LH servo valve. Flow from 2LD shall increase to full bench flow. Flow from 2RD shall not vary.
(7) Remove power from LH servo valve.
(8) Apply +20 ma max. DC power to the red wire on the RH servo valve. Flow from 2RD shall increase to full bench flow. Flow from 2LD shall not vary.
(9) Remove power from the RH servo valve.
(10) Apply +20 ma max. DC power to the green wire on the LH servo valve. Flow shall cease at 2LD and full bench flow shall begin at 1LU. Flow from 2RD shall not vary.
(11) Remove power from the LH servo valve.
(12) Apply +20 ma max. DC power to the green wire on the RH servo valve. Flow from 2RD shall cease and full bench flow shall result from 1RU. Flow from 2LD shall not vary.
(13) Remove pressure from assembly and power from servo valve. Plug ports until assembly on aircraft.



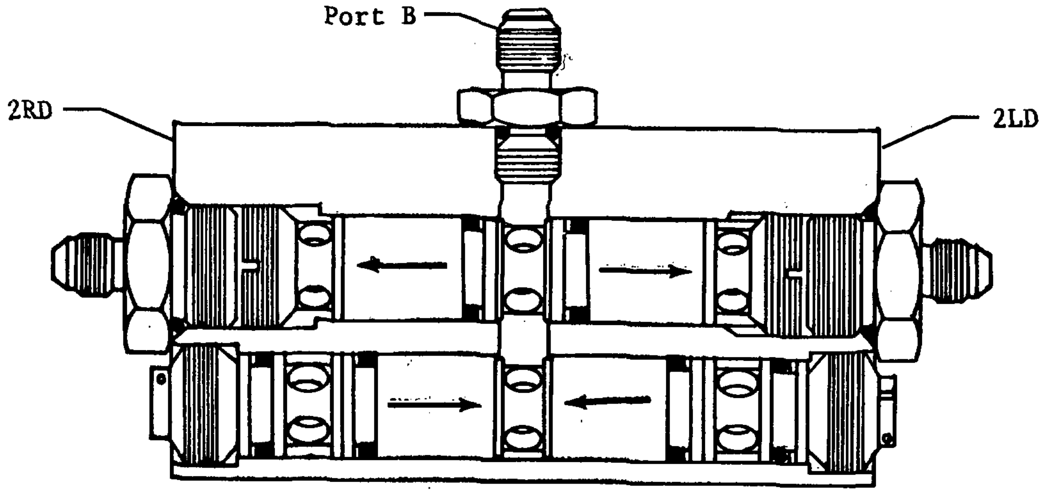
Spoiler Servo Valve Manifold Assembly
Figure 201 (Sheet 1 of 2)



Spoiler Servo Valve Manifold Assembly
Figure 201 (Sheet 1 of 2)

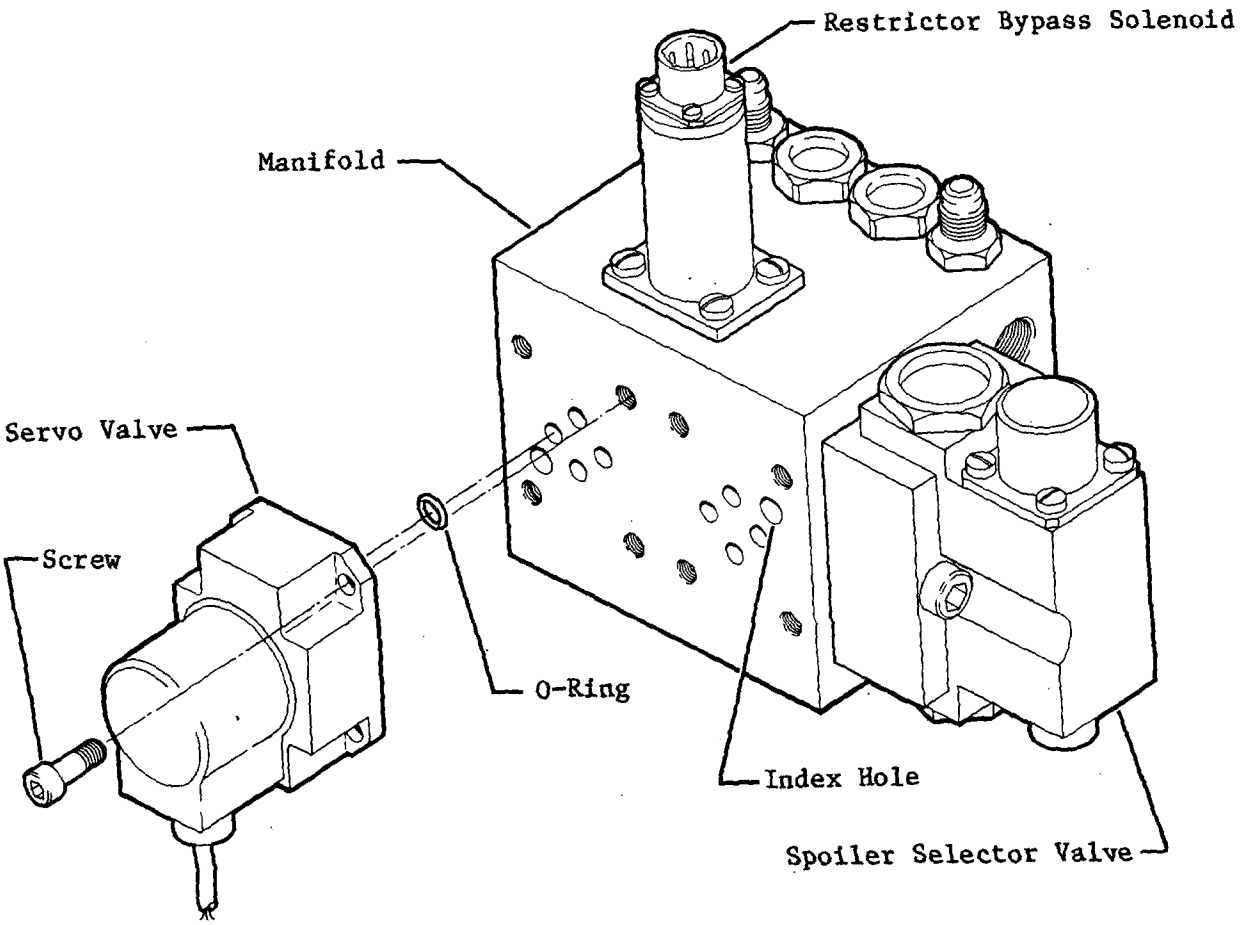


TOP VIEW



BOTTOM VIEW

Spoiler Servo Valve Manifold Assembly
Figure 201 (Sheet 2 of 2)



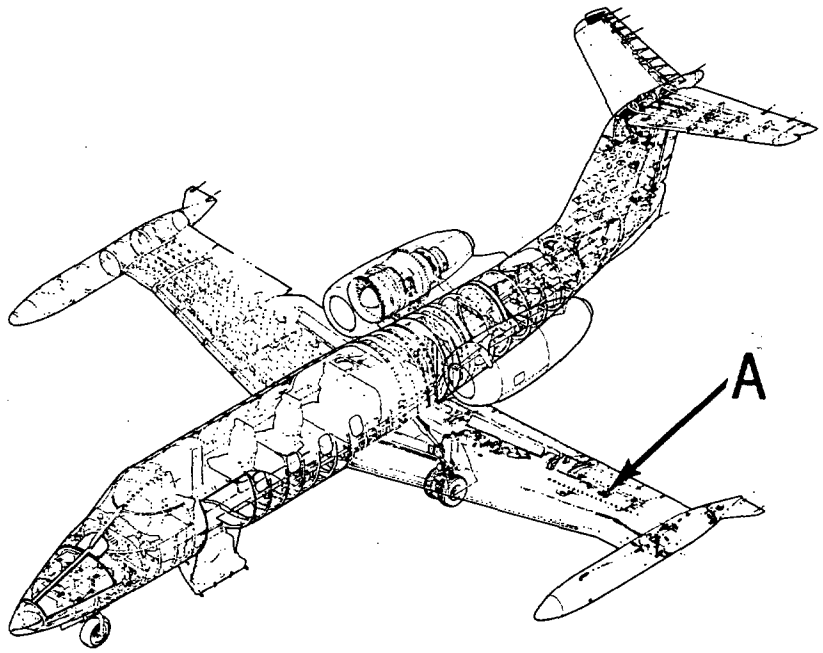
Spoiler Servo Valve Manifold
Figure 202

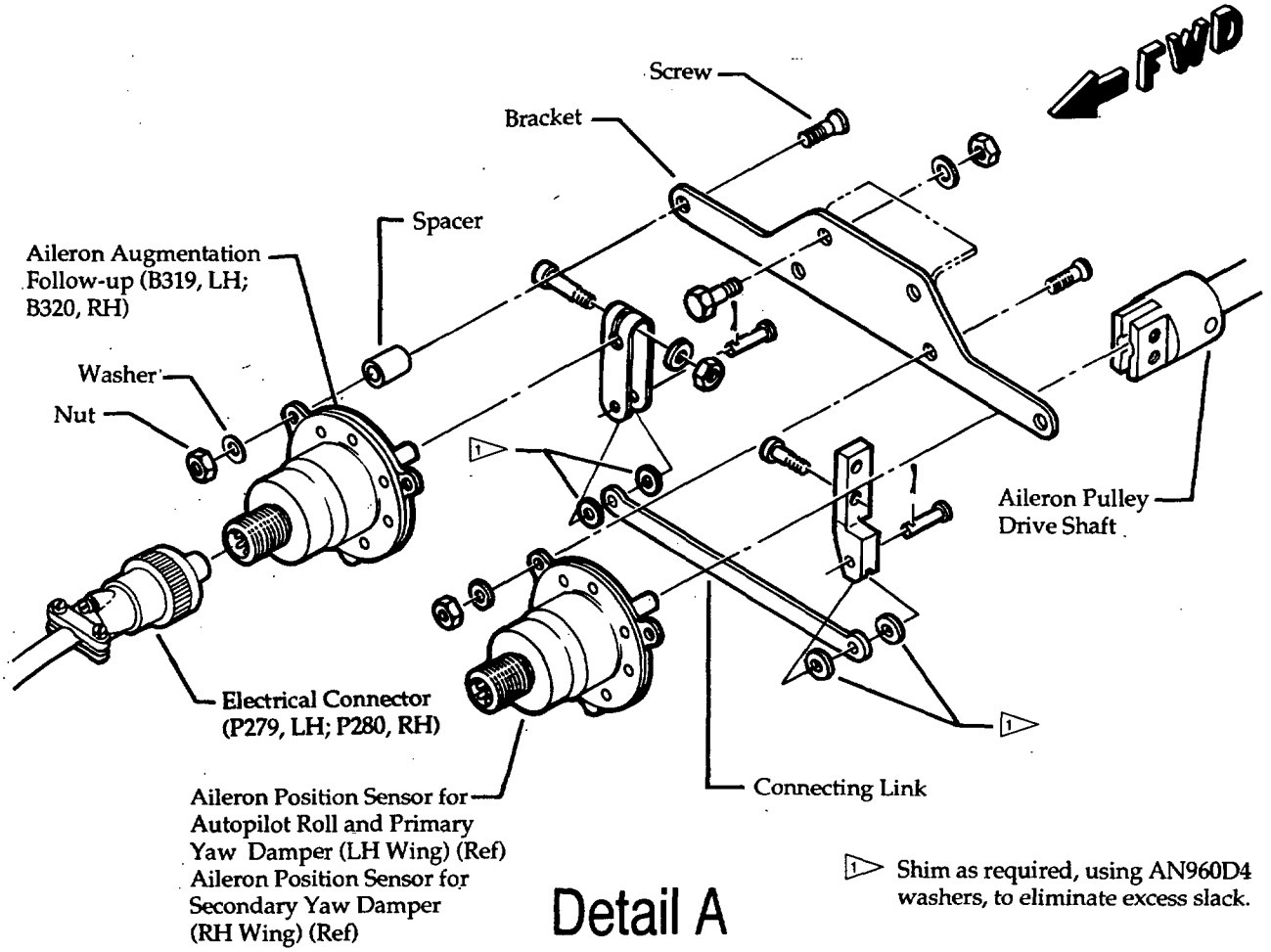
AILERON AUGMENTATION FOLLOW-UP - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Aileron Augmentation Follow-up (B319, LH; B320, RH) (See Figure 201.)
   (1) Remove wing access cover to gain access to aileron augmentation follow-up. (Refer to Chapter 57.)
   (2) Disconnect electrical connector (P279, LH; P280, RH) from follow-up.
   (3) Loosen attaching parts and disconnect link from aileron augmentation follow-up shaft.
   (4) Remove attaching parts and follow-up from bracket.

B. Install Aileron Augmentation Follow-up (B319, LH; B320, RH) (See Figure 201.)
   (1) Install follow-up and secure with attaching parts.
   (2) Install link on follow-up shaft.
   (3) Align red dots on follow-up shaft and housing.
   (4) Tighten link attaching parts.
   (5) Connect electrical connector (P279, LH; P280, RH) to aileron augmentation follow-up.
   (6) Check aileron augmentation follow-up for proper operation. (Refer to Adjustment/Test, 27-60-00.)
   (7) Install wing access cover. (Refer to Chapter 57.)
   (8) Restore aircraft to normal.





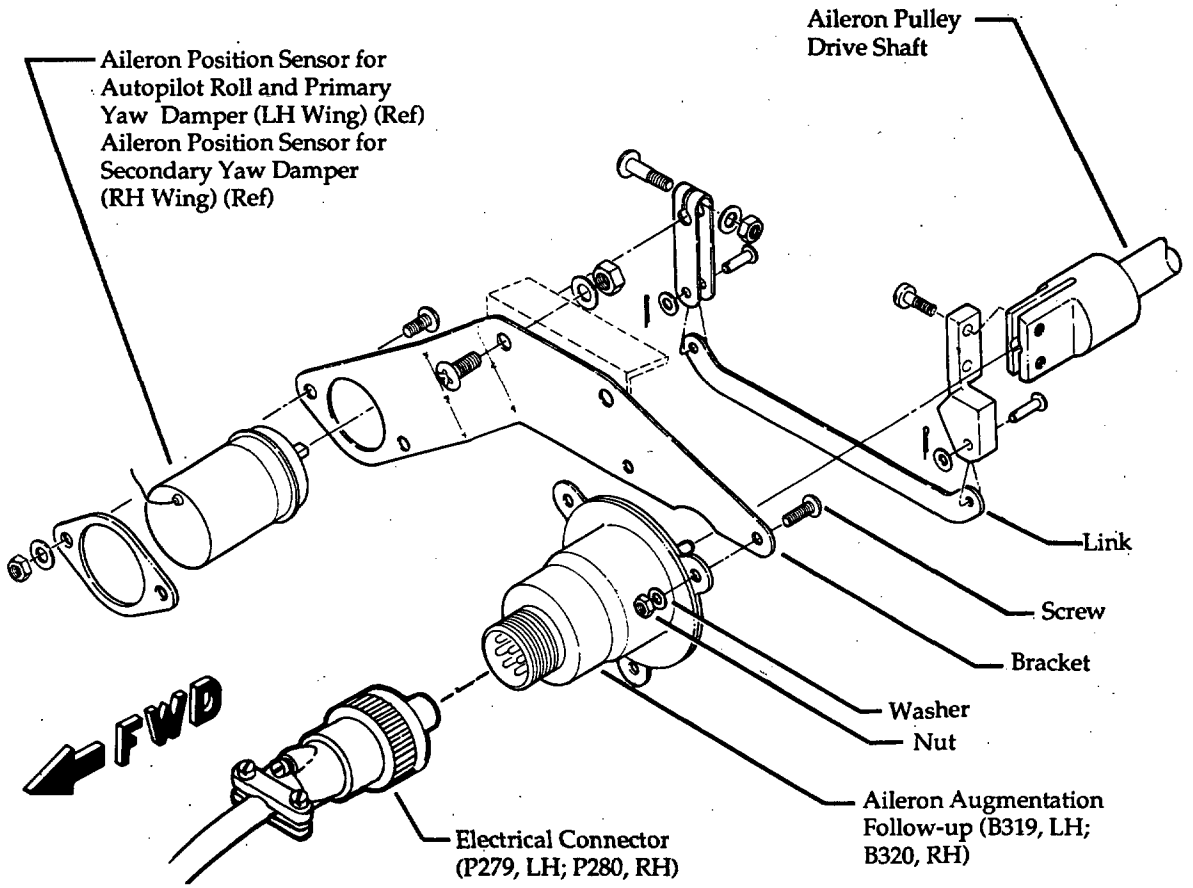
Detail A

8-10C

Aileron Augmentation Follow-up Installation
Figure 201 (Sheet 1 of 2)

EFFECTIVITY: AIRCRAFT 35-002 THRU 35-505 EXCEPT 35-408, 36-002 THRU 36-053 NOT MODIFIED PER AAK 83-2, INSTALLATION OF FC-530 AUTOPILOT

MM-99



Detail A

8-10C-2

Aileron Augmentation Follow-up Installation
Figure 201 (Sheet 2 of 2)

TEMPORARY REVISION NO. 27-63

MANUALS AFFECTED:	Learjet 35/35A/36/36A Maintenance Manual.
FILING INSTRUCTIONS:	Insert adjacent to 27-60-05, Page 201, dated Nov 4/83, and retain until further notice.
REASON:	Revised spoiler followup removal/installation procedure.
INSTRUCTIONS:	Add revised procedures as follows:

SPOILER FOLLOWUP - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: Removal and installation procedures for both spoiler followups are identical.

A. Removal of the Spoiler Followup

(1) Connect hydraulic and electrical power to the aircraft.
(2) Fully extend the spoilers and the flaps.

WARNING: SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM AIRCRAFT WITH HYDRAUILIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELECTRICAL POWER.

(3) Remove hydraulic power from the aircraft.
(4) Remove electrical power from the aircraft.
(5) Remove the attaching parts the connect the spoiler actuator rod end to the spoiler actuator arm. Block the spoiler up.
(6) Remove the attaching parts that connect the followup shaft to the followup arm.
(7) Disconnect and tag the electrical wires from the spoiler followup.
(8) Remove the spoiler followup retainer and spoiler followup by removing the attaching parts.

B. Installation of the Spoiler Followup

(1) Insert spoiler followup shaft through mounting bracket into hole in followup arm.
(2) Position retainer on mounting bracket and secure with attaching parts.
(3) Install attaching parts which secure followup arm to followup shaft.
(4) Remove tags and connect the spoiler followup electrical wiring to aircraft wiring.
(5) Install attaching parts securing spoiler actuator rod end to actuator arm. Remove previously installed block from spoiler.
(6) Perform spoiler system adjustment. (Refer to Adjustment/Test.)
(7) Perform spoiler system functional test. (Refer to Inspection/Check, 27-60-00.)
(8) Restore electrical power to aircraft.
(9) Restore hydraulic power to aircraft.

SPOILER FOLLOWUP - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

NOTE: Removal and installation procedures for both spoiler followups are identical.

A. Remove Spoiler Followup (See figure 201.)

(1) With hydraulic and electrical power supplied to aircraft, fully extend spoilers and flaps. Block spoilers up.

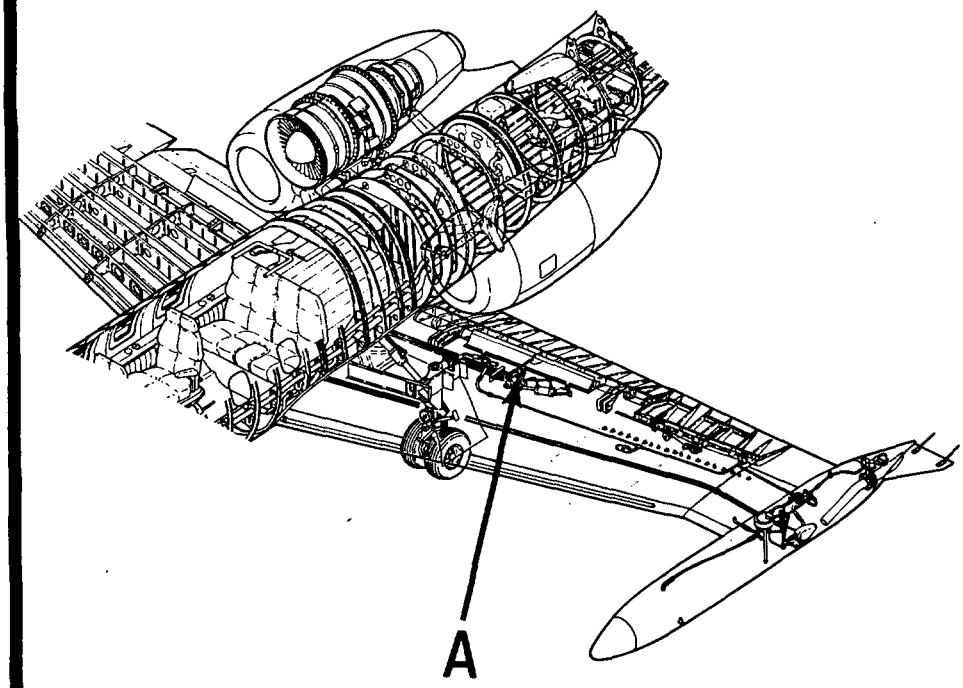
WARNING: ° TO PREVENT INADVERTENT SLAM DOWN, BLOCK SPOILERS WHEN EXTENDED.

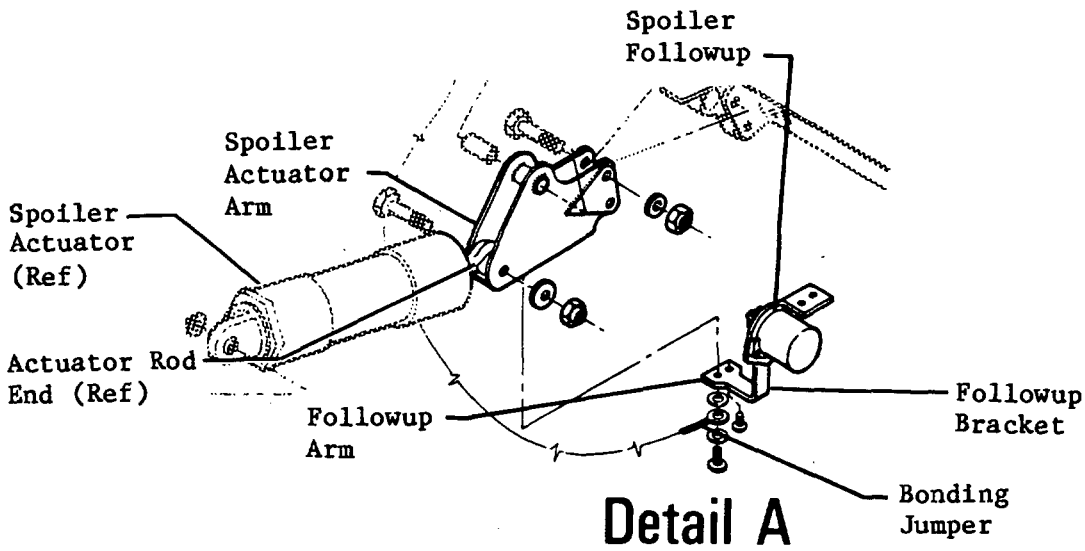
° SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM AIRCRAFT WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELECTRICAL POWER.

(2) Remove hydraulic power from aircraft.
(3) Remove electrical power from aircraft.
(4) Remove attaching parts which connect followup shaft to followup arm.
(5) Disconnect and tag electrical wires from spoiler followup.
(6) Remove spoiler followup retainer and spoiler followup by removing attaching parts.

B. Install Spoiler Followup (See figure 201.)

(1) Insert spoiler followup shaft through mounting bracket into hole in followup arm.
(2) Position retainer on mounting bracket and secure with attaching parts.
(3) Install attaching parts which secure followup arm to followup shaft.
(4) Remove tags and connect electrical wires to aircraft.
(5) Perform spoiler system adjustment. (Refer to Adjustment/Test.)
(6) Perform spoiler system functional test. (Refer to Inspection/Check, 27-60-00.)
(7) Restore electrical power to aircraft.
(8) Restore hydraulic power to aircraft.





(LH SIDE SHOWN)

Spoiler Followup Installation
Figure 201

4-32D-1W

2. ADJUSTMENT/TEST

A. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Spoileron Phase Detector/Test Box	2671007	Gates Learjet	Electrical phase check of spoiler followup.
AC Voltmeter (VTVM)		Commercially Available	General

B. Adjust Spoiler Followups

(1) Remove electrical power from aircraft.
(2) Gain access to spoiler computer.
(3) Disconnect electrical connector from spoiler computer.
(4) Connect Spoileron Phase Detector/Test Box test leads between spoiler computer electrical connector and spoiler computer.
(5) Restore electrical power to aircraft.
(6) Set Battery Switches to BAT 1 and BAT 2 and set Inverter Switches on.

WARNING: DO NOT APPLY HYDRAULIC POWER TO AIRCRAFT.

(7) Connect AC voltmeter between right spoiler VOLTS and GROUND terminals on test box.
(8) Ensure that spoiler is in full retract position and ailerons are centered.
(9) Loosen right spoiler followup attaching parts and rotate right spoiler followup until a reading of 0.8 vac is obtained on voltmeter; green (in-phase) RIGHT SPOILER FU light will illuminate.
(10) Manually extend spoiler, green (in-phase) RIGHT SPOILER FU light shall extinguish and red (out-of-phase) RIGHT SPOILER FU light shall illuminate.

NOTE: If followup phasing is not correct; red (out-of-phase) will not illuminate. Rotate followup housing 180°, and recheck for 0.8 vac in steps (8) thru (10) at correct phasing.

(11) Secure right spoiler followup.
(12) Set Battery Switches and Inverter Switches to OFF.
(13) Disconnect AC voltmeter from right spoiler VOLTS terminal and connect AC voltmeter to left spoiler VOLTS terminal.
(14) Ensure that left spoiler is in full retract position and ailerons are centered.

(15) Set Battery Switches to BAT 1 and BAT 2 and set Inverter Switches on.
(16) Loosen left spoiler followup attaching parts and rotate followup housing until a reading of 0.8 vac is obtained on voltmeter and green LEFT SPOILER FU light will illuminate.
(17) Manually extend spoiler; green (in-phase) light shall remain illuminated.

NOTE: If followup phasing is not correct; red (out-of-phase) light shall illuminate. If red (out-of-phase) light illuminates, rotate followup housing 180° and perform steps (16) and (17).

(18) Secure left spoiler followup.
(19) Set Battery Switches and Inverter Switches to OFF.
(20) Disconnect test box from system and connect electrical connector to spoiler computer.
(21) Install equipment removed to gain access to spoiler computer.

TEMPORARY REVISION NO. 27-64

MANUALS AFFECTED:	Learjet 35/35A/36/36A Maintenance Manual.
FILING INSTRUCTIONS:	Insert adjacent to 27-60-06, Page 201, dated Sep 25/92, and retain until further notice.
REASON:	Revised spoiler up-limit switch removal/installation procedure.
INSTRUCTIONS:	Add revised procedures as follows:

SPOILER UP-LIMIT SWITCH - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: Removal and installation procedures for both spoiler up-limit switches are identical.

A. Removal of the Spoiler up-limit switch (See Figure 201.)

(1) Acquire the necessary tools and equipment.

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Clinometer	2471004-1	Learjet Inc. Wichita, KS	Measuring angles
Multimeter	Model 260-7	Simpson	Resistance Measurment
Safety Wire		Commercially Available	Safetying Parts

(2) Connect hydraulic and electrical power to the aircraft,

(3) Fully extend the spoilers and flaps.

WARNING: SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM AIRCRAFT WITH HYDRAUILIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELCTRICAL POWER.

(4) Remove hydraulic power from the aircraft.

(5) Remove electrical power from the aircraft.

(6) Remove the attaching parts the connect the spoiler actuator rod end to the spoiler actuator arm. Block the spoiler up.

(7) Disconnect and tag the electrical wiring at the spoiler up-limit switch.

(8) Remove safety wire and attaching parts securing the spoiler up-limit switch to mounting bracket.

B. Installation of the Spoiler Up-Limit Switch (See Figure 201.)

(1) Position spoiler up-limit switch on the mounting bracket and secure with attaching parts.

NOTE: Do not safety wire spoiler up-limit switch until after switch adjustment has been performed.

(2) Install a clinometer on the spoiler's upper surface and adjust the spoiler to 16°; secure in position.
(3) Using an ohmmeter, probe between the spoiler up-limit switch leads. Adjust spoiler up-limit switch so that the continuity is broken at 16° (+1/-0°).
(4) Verify that spoiler up-limit switch is adjusted properly by manually raising and lowering spoiler. Verify continuity when the spoiler is below 16° and that an open circuit exists at 16° (+1/-0°).
(5) After spoiler up-limit switch is properly adjusted, secure the spoiler up-limit switch attaching parts and install safety wire.
(6) Remove the tags and connect the electrical wiring to spoiler up-limit switch.
(7) Position spoiler actuator rod end to the actuator arm and install attaching parts. Remove the previously installed block from the spoiler.

SPOILER UP-LIMIT SWITCH - MAINTENANCE PRACTICES

1. Tools and Equipment

NOTE: Equivalent substitutes may be used in lieu of the following:

NAME	PART NUMBER	MANUFACTURER	USE
Clinometer	2471004-1	Learjet Inc. Wichita, KS	Measuring Angles
Multimeter	Model 260-7	Simpson	Resistance Measurement
Safety Wire		Commercially Available	Safetying Parts

2. Removal/Installation

NOTE: Removal and installation procedures for both spoiler up-limit switches are identical.

A. Remove Spoiler Up-Limit Switch (See figure 201.)
(1) With hydraulic and electrical power supplied to aircraft, fully extend spoilers and flaps. Block spoilers up.

WARNING:
- **TO PREVENT INADVERTENT SLAM DOWN, BLOCK SPOILERS WHEN EXTENDED.**
- **SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM AIRCRAFT WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELECTRICAL POWER.**

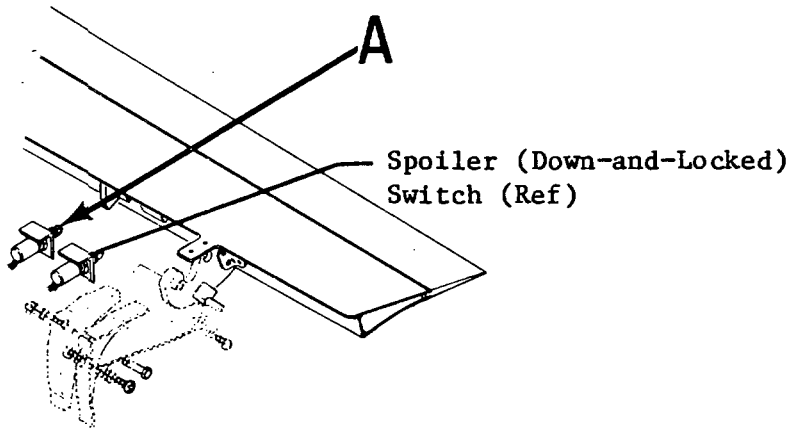
(2) Remove hydraulic power from aircraft.
(3) Remove electrical power from aircraft.
(4) Remove attaching parts securing spoiler actuator rod end to spoiler actuator arm.
(5) Disconnect and tag electrical wiring at spoiler up-limit switch.
(6) Remove safety wire and attaching parts securing spoiler up-limit switch to mounting bracket.

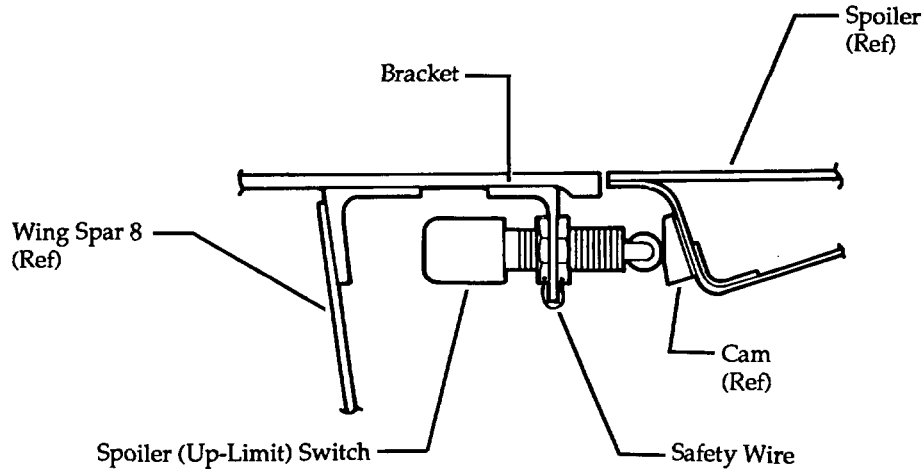
B. Install Spoiler Up-Limit Switch (See figure 201.)
(1) Position spoiler up-limit switch on mounting bracket and secure with attaching parts.

NOTE: Do not safety wire spoiler up-limit switch until after switch adjustment has been performed.

(2) Install a clinometer on spoiler upper surface and adjust spoiler to 16°; secure in position.
(3) Using an ohmmeter, probe between spoiler up-limit switch leads. Adjust spoiler up-limit switch so that continuity is broken at 16° (+1°/-0°).
(4) Verify that spoiler up-limit switch is adjusted properly by manually raising and lowering spoiler. Verify continuity when spoiler is below 16° and that an open circuit exists at 16° (+1°/-0°).
(5) After spoiler up-limit switch is properly adjusted, secure spoiler up-limit switch attaching parts and install safety wire.
(6) Remove tags and connect electrical wiring to spoiler up-limit switches.

(7) Install spoiler actuator rod end to spoiler actuator arm and secure with attaching parts. Remove blocks from spoilers.
(8) Perform operational check of spoiler system. (Refer to Inspection/ Check, 27-60-00.)
(9) Restore electrical power to aircraft.
(10) Restore hydraulic power to aircraft.
(11) Retract flaps and spoilers in full retract position.





Detail A

Spoiler Up-Limit Switch Installation
Figure 201

TEMPORARY REVISION NO. 27-65

MANUALS AFFECTED: Learjet 35/35A/36/36A Maintenance Manual.

FILING INSTRUCTIONS: Insert adjacent to 27-60-07, Page 201, dated Nov 4/83, and retain until further notice.

REASON: Revised spoiler down-and-locked switch removal/installation procedure.

INSTRUCTIONS: Add revised procedures as follows:

SPOILER DOWN-AND-LOCKED SWITCH - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: Removal and installation procedures for both spoiler down-and-locked switches are identical.

A. Removal of the Spoiler Down-and-Locked Switch (See Figure 201.)
  (1) Connect hydraulic and electrical power to the aircraft.
  (2) Fully extend the spoilers and the flaps.

WARNING: SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM AIRCRAFT WITH HYDRAUILIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELECTRICAL POWER.

  (3) Remove hydraulic power from the aircraft.
  (4) Remove electrical power from the aircraft.
  (5) Remove the attaching parts the connect the spoiler actuator rod end to the spoiler actuator arm. Block the spoiler up.
  (6) Disconnect and tag the electrical wires from the spoiler down-and-locked switch.
  (7) Remove safety wire and attaching parts securing spoiler down-and-locked switch to mounting bracket.

B. Installation of the Spoiler Down-and Locked Switch (See Figure 201.)
  (1) Position the spoiler down-and-locked switch on the mounting bracket and install the attaching parts.

  NOTE: Do not safety wire the spoiler down-and-locked switch until after the switch adjustment has been performed.

  (2) Using an ohmmeter, probe between spoiler down-and-locked switch leads. Adjust the spoiler down-and-locked switch so that continuity is broken only when the spoiler is in the down-and-locked position (0°).

(3) Make sure that the spoiler down-and-locked switch is adjusted properly by manually raising and lowering the spoiler. Make sure that there is continuity when the spoiler is above 0° and that an open circuit exists when the spoiler is in the down-and-locked position (0°).
(4) After the spoiler down-and-locked switch is properly adjusted, secure the spoiler down-and-locked switch attaching parts and install safety wire.
(5) Remove tags and connect electrical wiring to spoiler down-and-locked switch.
(6) Install the spoiler actuator rod end to the spoiler actuator arm and secure with the attaching parts. Remove the previously installed block from the spoiler.

SPOILER DOWN-AND-LOCKED SWITCH - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

NOTE: Removal and installation procedures for both spoiler down-and-locked switches are identical.

A. Remove Spoiler Down-and-Locked Switch (See figure 201.)

(1) With hydraulic and electrical power supplied to aircraft, fully extend spoilers and flaps. Block spoilers up.

WARNING: ° TO PREVENT INADVERTENT SLAM DOWN, BLOCK SPOILERS WHEN EXTENDED.

° SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM AIRCRAFT WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST, AND THEN REMOVE ELECTRICAL POWER.

(2) Remove hydraulic power from aircraft.
(3) Remove electrical power from aircraft.
(4) Remove attaching parts securing spoiler actuator rod end to spoiler actuator arm.
(5) Disconnect and tag electrical wiring at spoiler down-and-locked switch.
(6) Remove safety wire and attaching parts securing spoiler down-and-locked switch to mounting bracket.

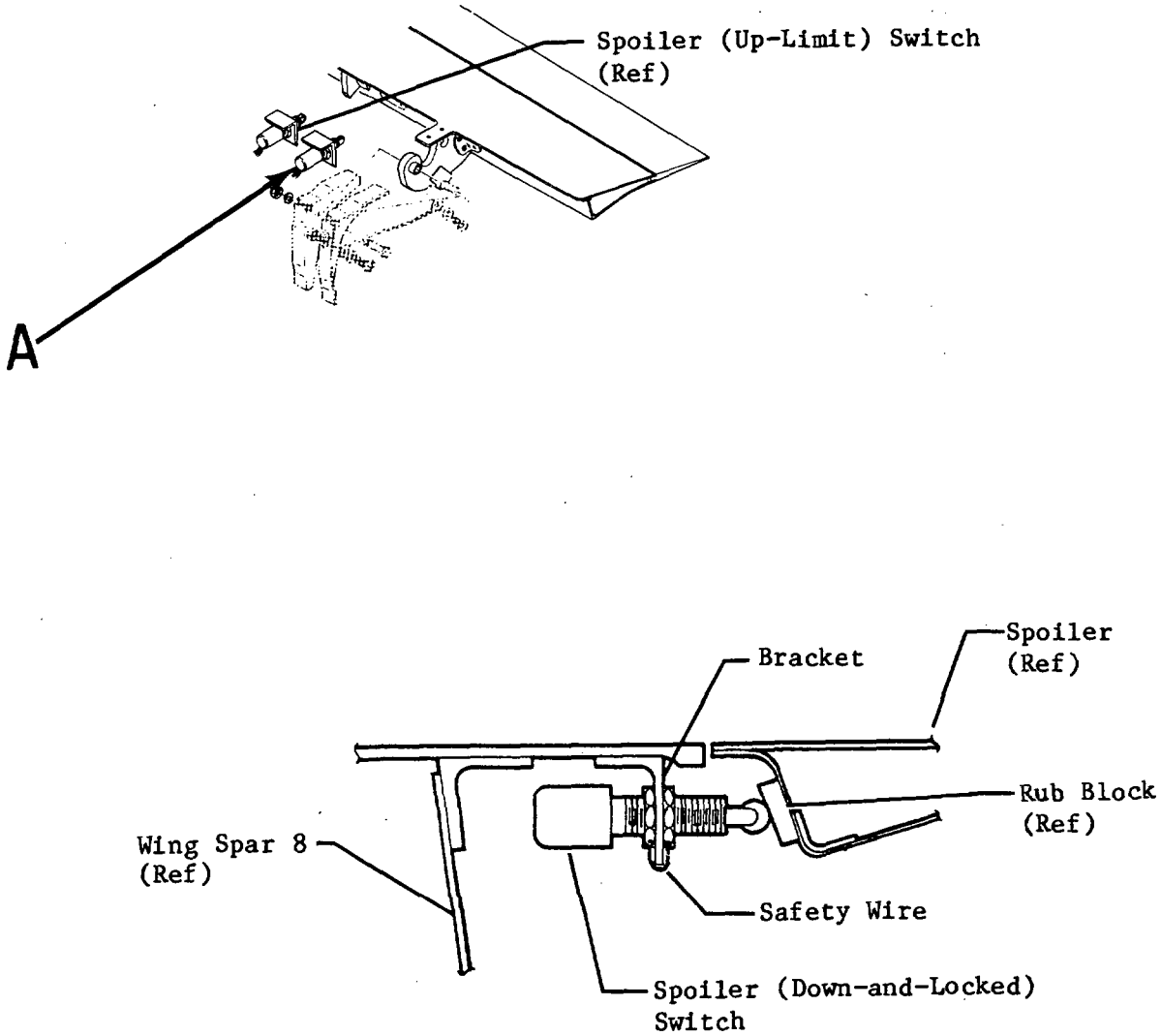
B. Install Spoiler Down-and-Locked Switch (See figure 201.)

(1) Position spoiler down-and-locked switch on mounting bracket and secure with attaching parts.

NOTE: Do not safety wire spoiler down-and-locked switch until after switch adjustment has been performed.

(2) Using an ohmmeter, probe between spoiler down-and-locked switch leads. Adjust spoiler down-and-locked switch so that continuity is broken only when spoiler is in down-in-locked position (0°).
(3) Verify that spoiler down-and-locked switch is adjusted properly by manually raising and lowering spoiler. Verify continuity when spoiler is above 0° and that an open circuit exists when spoiler is in down-and-locked position (0°).
(4) After spoiler down-and-locked switch is properly adjusted, secure spoiler down-and-locked switch attaching parts and install safety wire.
(5) Remove tags and connect electrical wiring to spoiler down-and-locked switch.
(6) Install spoiler actuator rod end to spoiler actuator arm and secure with attaching parts. Remove blocks from spoilers.

(7) Perform operational check of spoiler system. (Refer to Inspection/ Check, 27-60-00.)
(9) Restore electrical power to aircraft.
(9) Restore hydraulic power to aircraft.
(10) Stow flaps and spoilers in full retract position.



Spoiler Down-and-Locked Switch Installation
Figure 201

SPOILERON COMPUTER - MAINTENANCE PRACTICES

1. Removal/Installation

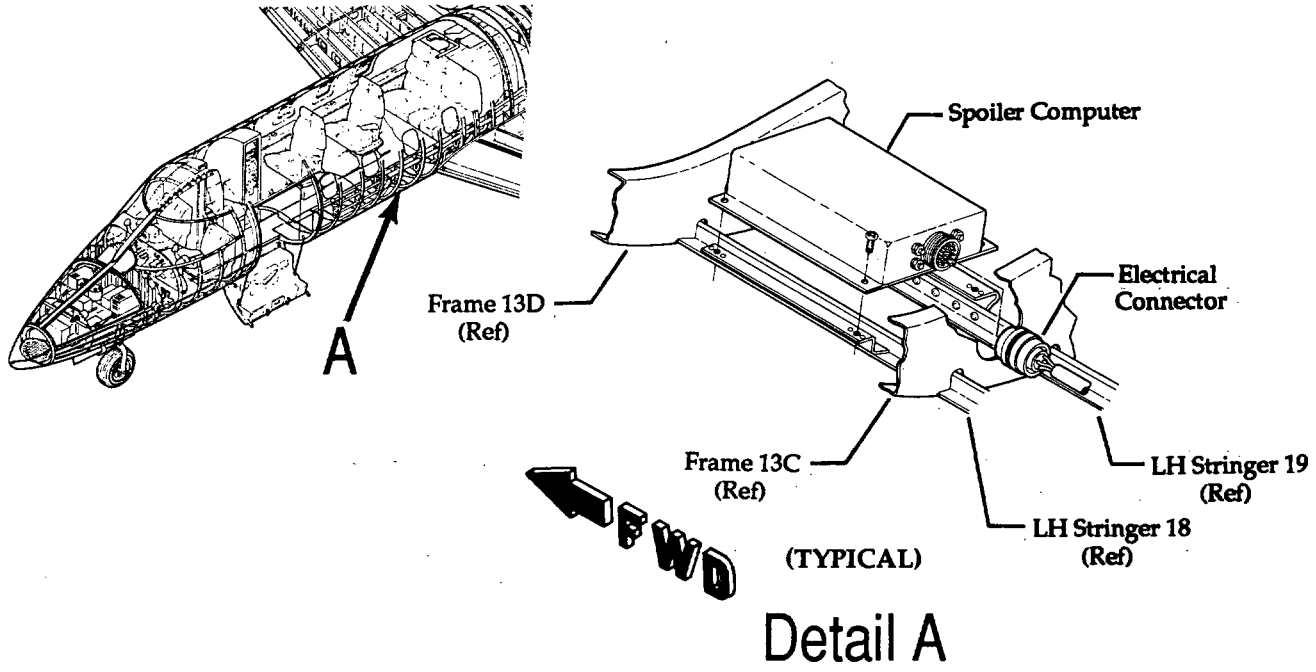
NOTE: *On Aircraft 35-002 thru 35-065, except 35-050, and 36-002 thru 36-017*, the spoileron computer is located beneath the floorboards at FS 332 and LBL 10. *On Aircraft 35-050, 35-066 thru 35-296 and 36-018 thru 36-044*, the spoileron computer is located at FS 286 and LBL 12. *On Aircraft 35-297 and Subsequent and 36-045 and Subsequent*, the spoileron computer is located at FS 310 and LBL 12. *On Aircraft equipped with camera installation*, the spoileron computer is located at FS 275 and LBL 12.

A. Remove Spoileron Computer (See figure 201.)
   (1) Remove electrical power from aircraft.
   (2) Remove applicable cabin furnishings and equipment as required to gain access to spoileron computer.
   (3) Disconnect electrical connector from computer.
   (4) Remove attaching parts securing computer to aircraft structure; remove computer.

B. Install Spoileron Computer (See figure 201.)
   (1) Position spoileron computer at its appropriate location on aircraft structure and secure with attaching parts.

   NOTE: Check electrical resistance between spoileron computer and aircraft structure. Resistance shall NOT be greater than value specified in Chapter 20 of the Wiring Manual.

   (2) Connect electrical connector to computer.
   (3) Perform functional test of spoiler system. (Refer to 27-60-00, Adjustment/Test.)
   (4) Install previously removed equipment and furnishings.
   (5) Restore electrical power to aircraft.
   (6) Restore aircraft to normal.



A9-219B

Spoileron Computer Installation
Figure 201

EFFECTIVITY: ALL