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(Formerly AN 01-90CDB-3)

HANDBOOK
STRUCTURAL REPAIR INSTRUCTIONS

USAF SERIES

C-45G
TC-45G
C-45H

AIRCRAFT

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INTRODUCTION

This structural repair handbook is issued to provide assistance and instruction to those who make structural repairs to the C-45G and the TC-45G airplanes. In order to become thoroughly familiar with the basic fundamentals of repairs set forth in this handbook, Section I should be read in its entirety. Section I provides information of a general nature which will be used in conjunction with the more detailed description of repairs in the latter sections.

In Section II, a comprehensive outline of the three wing sections is given, with instructions for repair ranging from the patching of skin damage to replacement of the outboard center section truss assembly.

Section III contains information for the repair of structural damage to members of the tail group while Section IV, V, and VI cover in necessary detail the repairs of the remaining components of the airframe, with the exception of the fabric covered control surfaces which are discussed in Section VII.

An extrusion chart showing suitable substitutes comprises Section VIII.

Also provided as part of this handbook is an illustrated appendix of typical repairs, and tables and charts of repair materials.

SECTION I

GENERAL

TABLE OF CONTENTS

1-1. TYPE OF CONSTRUCTION.

1-2. The C-45G and the TC-45G airplanes employ a semi-monocoque construction, wherein stresses are carried by both the skin and the stiffener members. Due to this interrelationship, it is necessary that adequate stress connections be maintained at all times whenever one part is bolted or riveted to another, to insure the proper distribution of loads. Loads are both introduced and distributed by bulkheads, spars, and stringers. This results occasionally in local stress concentrations in the skin, which must be carefully dispersed. Examination of rivet patterns at any structural joint (including skin splices) will provide indication of the extent of stress concentration at this point and will serve as a guide when replacing damaged portions of the structure.

1-3. The fuselage framework of the airplane consists of bulkheads and longitudinal stringers to which the outside skin of the airplane is riveted. A highly heat-treated, welded steel tube truss carries the main wing spar stresses, and also provides supporting structure for the engines and landing gear. The wing stubs are not demountable.

1-4. INVESTIGATING DAMAGE.

1-5. In investigating the extent of damage, especially damage from shock, it is necessary to make an extensive inspection of the structure. A severe shock can be transmitted from one structural member to another and may result in damage or cause distortion of members remote from the point of impact. Wrinkled skin or fabric, distorted rivets and elongated or damaged bolt or rivet holes are all indications of damage to adjacent structures.

1-6. SUPPORT OF STRUCTURE DURING REPAIR.

1-7. During repair, it is necessary that the airplane be adequately supported in order that distortion or injury will not result. It is essential that the area to be repaired is supported solidly to maintain a firm and correct position with the surrounding area during repair. Measurements made from outstanding points near the damaged area before and after repair provide a simple means of checking for distortion. Fuel and oil tanks should be emptied before supporting the airplane for major repairs. The airplane will be supported by jacks, hoists, supports, or jigs.

WARNING

When the tail group is removed for a repair or replacement, the center of gravity shifts forward and the load is redistributed to the support points. Failure to sandbag the aft section or secure it to support structure would allow the center of gravity to shift forward and upset the aircraft.

1-8. JACKING.

1-9. The repair of many injuries requires that the aircraft be raised off the ground. The main jack points are located on the underside of the wing center

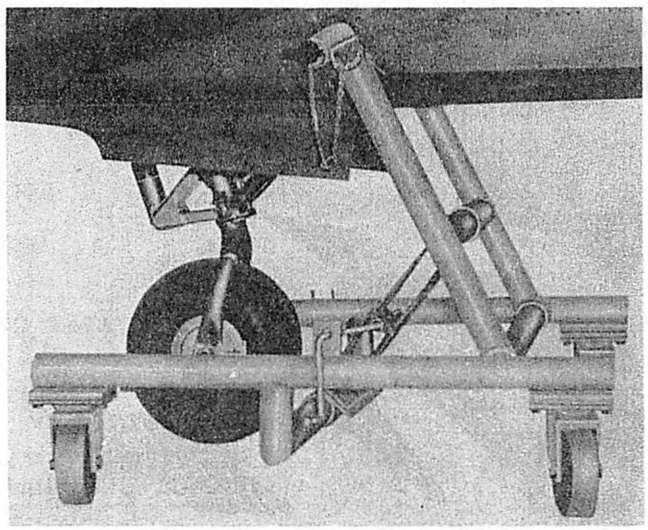


Figure 1-2. Tail Jacking

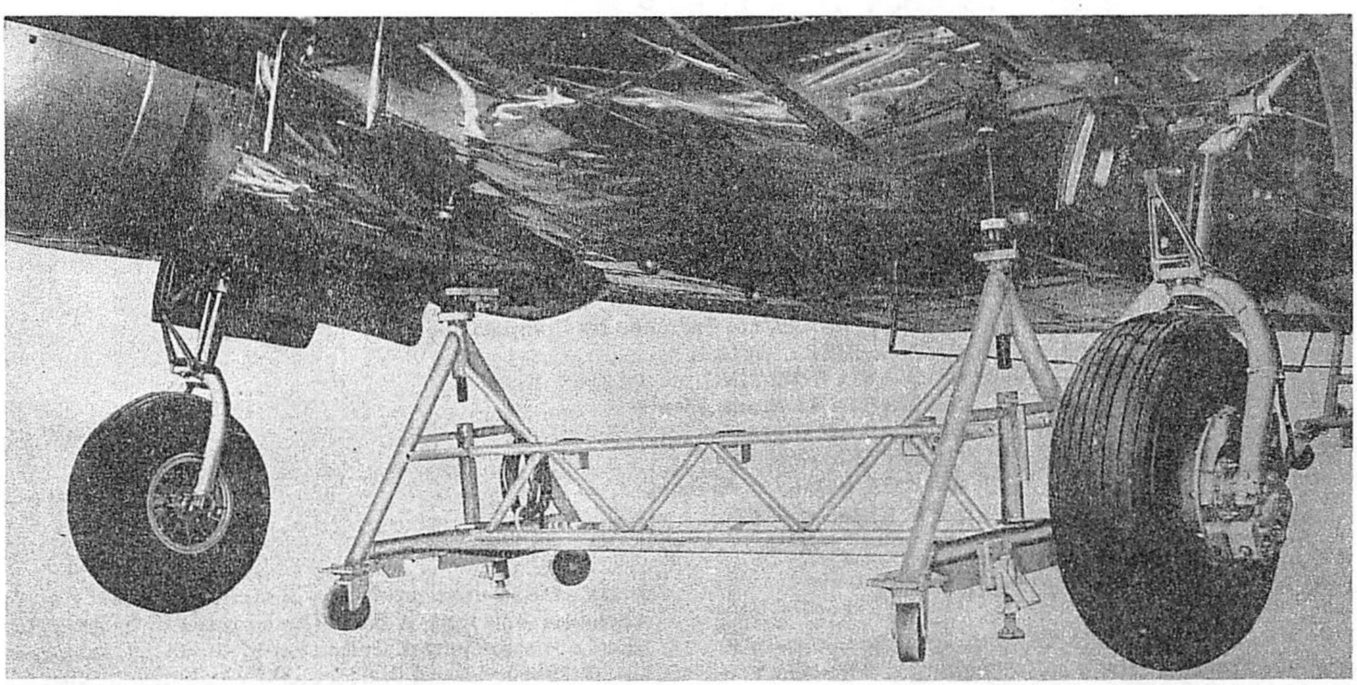


Figure 1-1. Main Gear Jacking

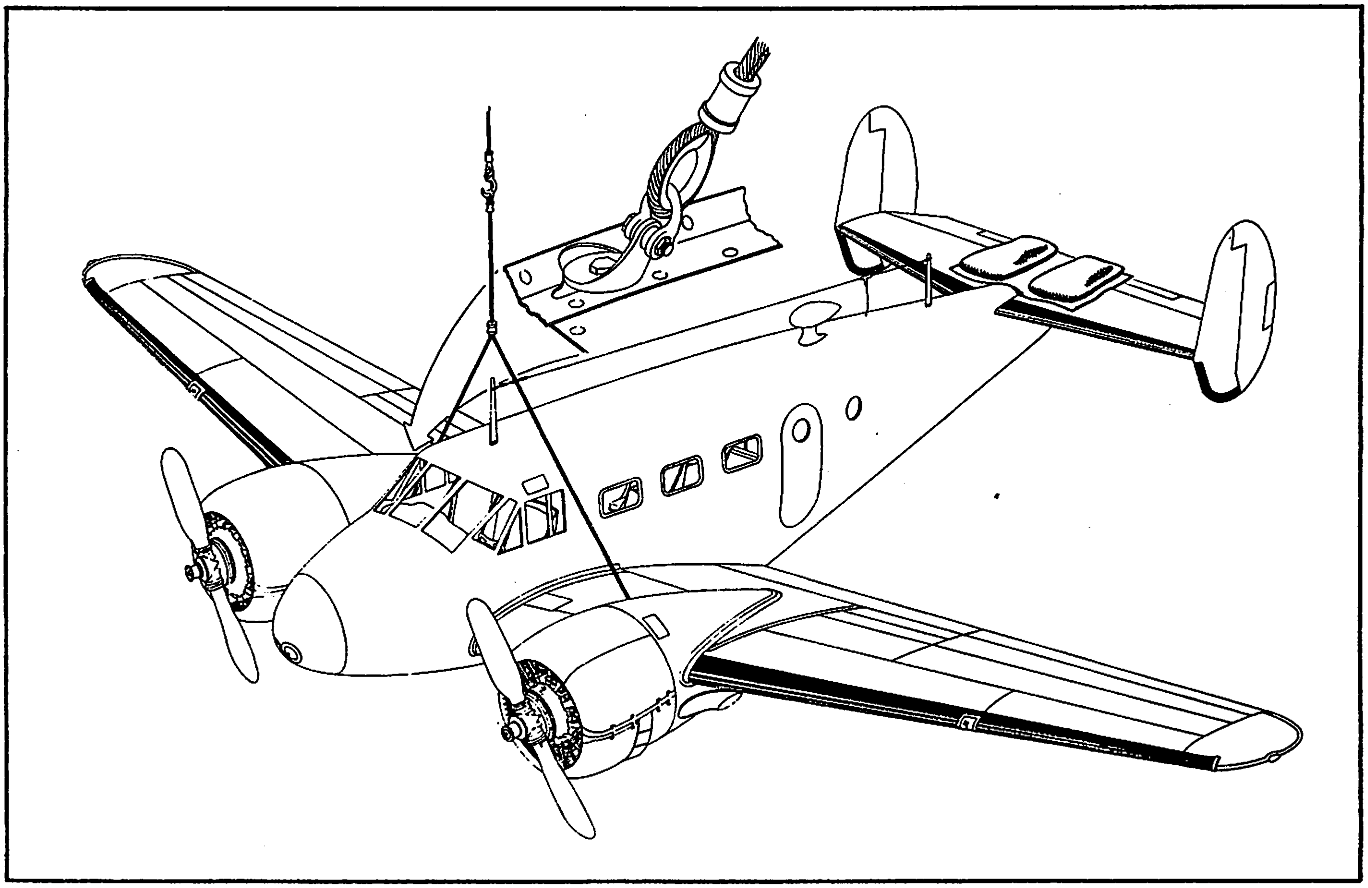


Figure 1-3. Hoisting Aircraft with One Hoist

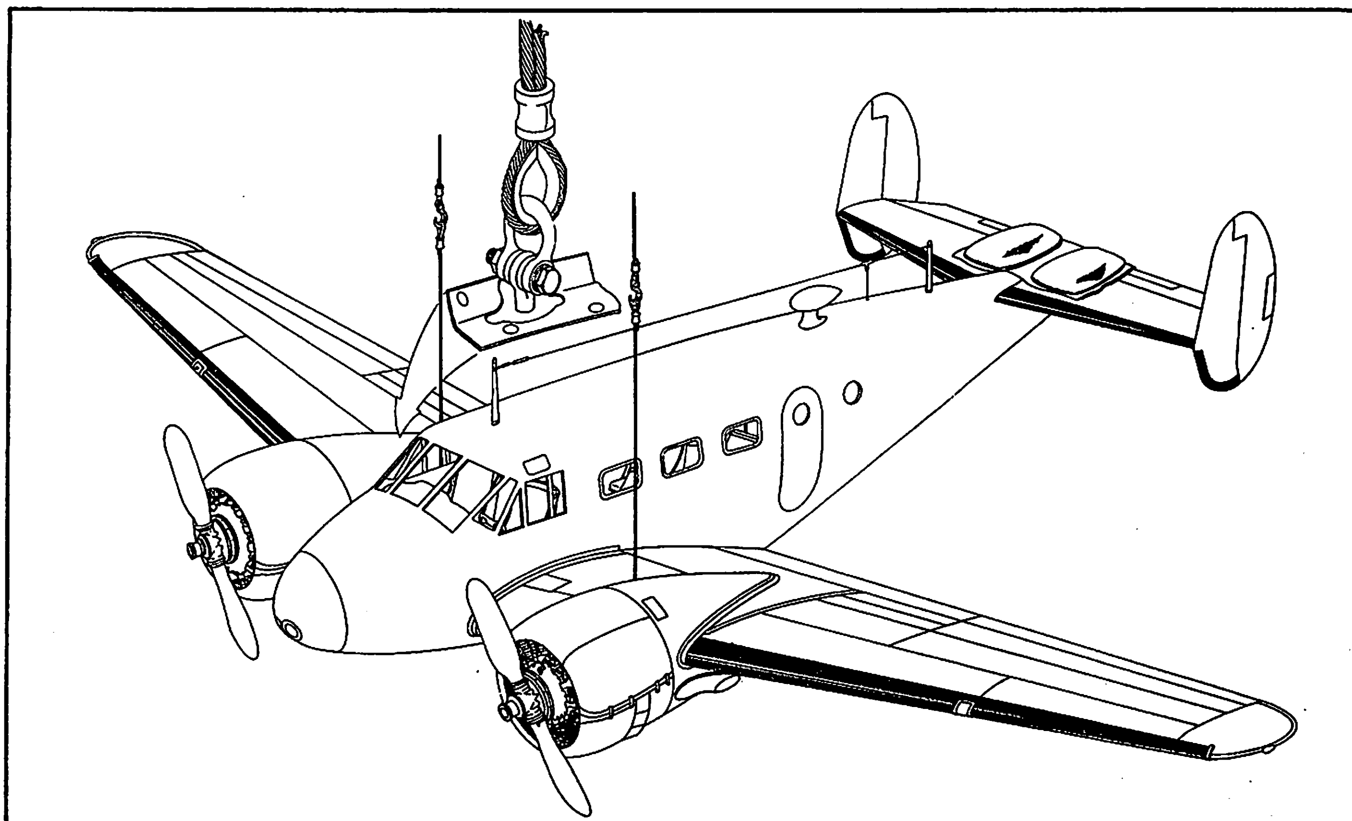


Figure 1-4. Hoisting Aircraft with Two Hoists

section between the fuselage and each nacelle. Three rubber Fork plugs must be removed and the jack pads (Part 84-180930) installed. These pads are supplied with each airplane. (See figure 1-1). Individual wheel jack points are provided on the inboard side of each main landing gear fork. The tail jack point is located just aft of the tail wheel. (See figure 1-2).

NOTE

When using jacks, be sure jack is suitable for jack point. When using wheel jacks, care must be exercised to prevent the aircraft from slipping from the jacks.

a. A jack should never be used near a damaged area where further injury might result.
b. Never leave equipment (ladders, benches, jigs, etc.) under a jacked aircraft while not in use. This precaution may prevent damage in case of a jack slip or failure.

1-10. HOISTING.

1-11. Points for hoisting the aircraft are located on the upper wing stub just inboard of the upper rear nacelle. Access to the fittings is gained by removing the fabric patch at these points. If one hoist (of at least 5-ton capacity) is used, bolt the attaching brackets to the truss. The chains and cables used in single point suspension should clear the cabin approximately 2 inches so that a direct pull will be placed on the lugs. (See figure 1-3.) This will prevent the lugs from being bent.

WARNING

If two hoists (of at least 3-ton capacity) are used, the vertical lift eye bolts are used. (See figure 1-4). When hoisting, approximately 200 pounds of weight must be applied to the tail to keep the aircraft from nosing over.

1-12. LIFTING.

1-13. The tail of the aircraft may be lifted by inserting a steel bar (maximum diameter - 3/4 inch) through the "lift" holes (located in the fuselage just aft of Bulkhead 13.

CAUTION

Do not attempt to raise the tail of the aircraft by lifting against the horizontal stabilizer as the ribs and stringers maybe seriously damaged.

1-14. LOCATION OF LEVELING POINTS.

1-15. Leveling points for longitudinal leveling are located on top of the fuselage, just forward of the cabin door. Place a straight edge on leveling points and level as shown in figure 1-5. Leveling points for lateral leveling are located on the underside of the fuselage at the center section truss. Place a straight edge across leveling points and level as shown in figure 1-6.

1-16. CLASSIFICATION OF DAMAGE.

1-17. NEGLIGIBLE DAMAGE. This includes damage not endangering the strength or function of the damaged area. Location of the injury determines to a large extent whether or not the damage is negligible. Nicks, dents, and scratches generally are considered negligible damage, except on certain highly-stressed members; i.e., landing gears. Damage of this nature on landing gears will require replacement of the assembly.

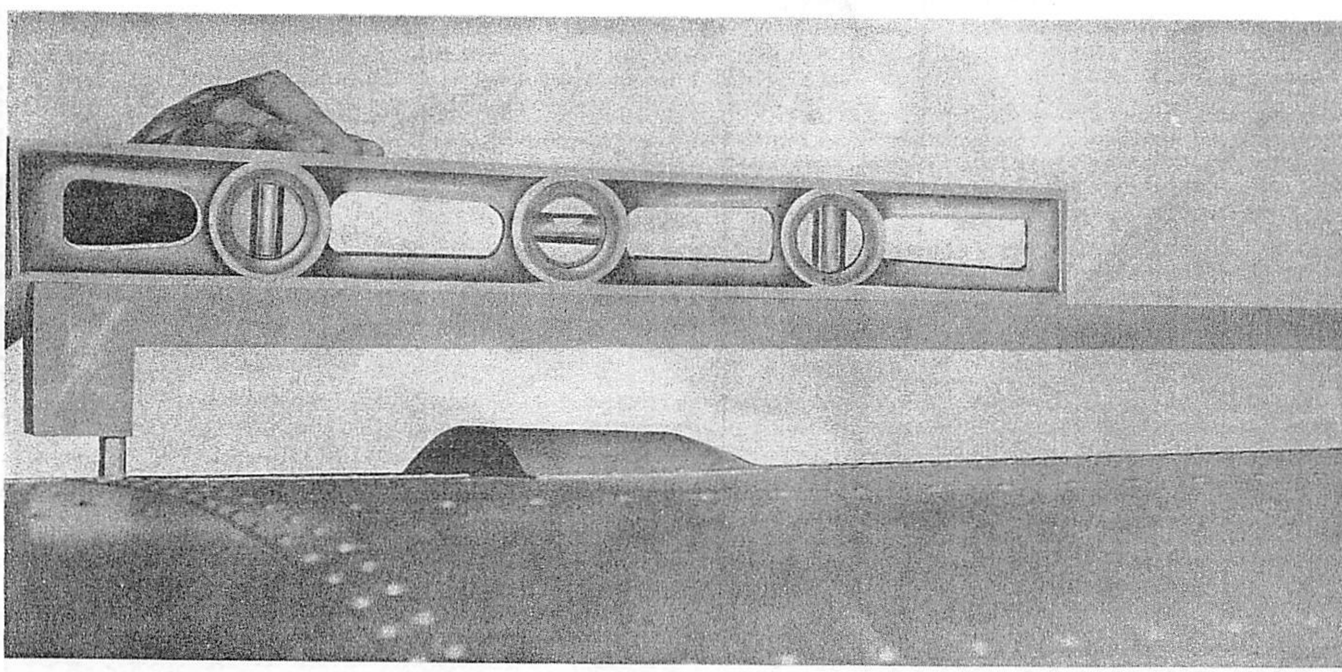


Figure 1-5. Longitudinal Leveling

1-18. DAMAGE REPAIRABLE BY PATCHING. This type of damage usually consists of punctures, cracks, or tears, of a unit such as skin, rib, or spar. Repair is made by the addition of material similar to that of the damaged area.

1-19. DAMAGE REPAIRABLE BY INSERTION. This type of damage is of such an extent that the damaged area must be removed, but the part as an entire unit need not be replaced. The damaged portion is removed and replaced by material of similar composition, shape, and size and secured in place by splices or patches which may cover the complete insert and overlap the original area.

1-20. DAMAGE NECESSITATING REPLACEMENT OF PARTS. This method shall be used only when other methods of repair are unsatisfactory or the reasons for not repairing the damaged area are adequate, for example: a damaged part is too small for repair; or in an inaccessible area; the type of material makes repair difficult; tools and material for proper repair are not available; or the part to be repaired is situated so that repair would weaken the area.

1-21. PREPARING DAMAGED AREAS FOR REPAIR. After examining and classifying a damaged area, it must be prepared for repair. This includes the removal of ragged edges, bends, tears, cracks, punctures and similar injuries. All such damages should be cut away with a file, hacksaw, or snips. Edges left by removal of the damaged area should be parallel to the unit edges when they are square or rectangular in shape. The corners of rectangular areas must be circular in shape. Smooth out all abrasions and dents.

1-22. TYPES OF REPAIR.

1-23. Structural repairs to the C-45G and the TC-45G airplanes will consist mainly of repairs to sheet-metal structures. The welded, steel-tube structure is for the most part, highly heat-treated. Indiscriminate welding of a frame, therefore, is prohibited. All welding repairs which may be accomplished without endangering the structural safety of the frame will be described in detail and should be rigidly observed. The use of wood in the construction and repair of the airplane is limited to nonstructural members, such as floorboards and upholstery tacking strips.

1-24. Determining the type of repair to be used on a certain damaged area involves ascertaining whether or not the damaged member is repairable. In many instances it is preferable to repair rather than replace to prevent grounding of the airplane for a long period of time while replacement is being made. Consideration also must be given the location of the damaged member accessibility, and function in the aircraft. Generally, it may be more desirable to make replacement if the damage was to highly stressed parts, which have peculiarities, such as a high degree of heat-treatment; special tolerances; or to such extent that replacement is obvious. However, it is sometimes impossible to repair damage by replacement, due to the unavailability of parts. In this case, substitutions or repairs made in accordance with the recommended procedure of this handbook will facilitate repairs and restore adequate strength to the damaged part.

NOTE

When a new repair involves an area immediately adjacent to or within a few inches of an already existing repair, it is better to remove the old repair and incorporate the entire area into the new repair.

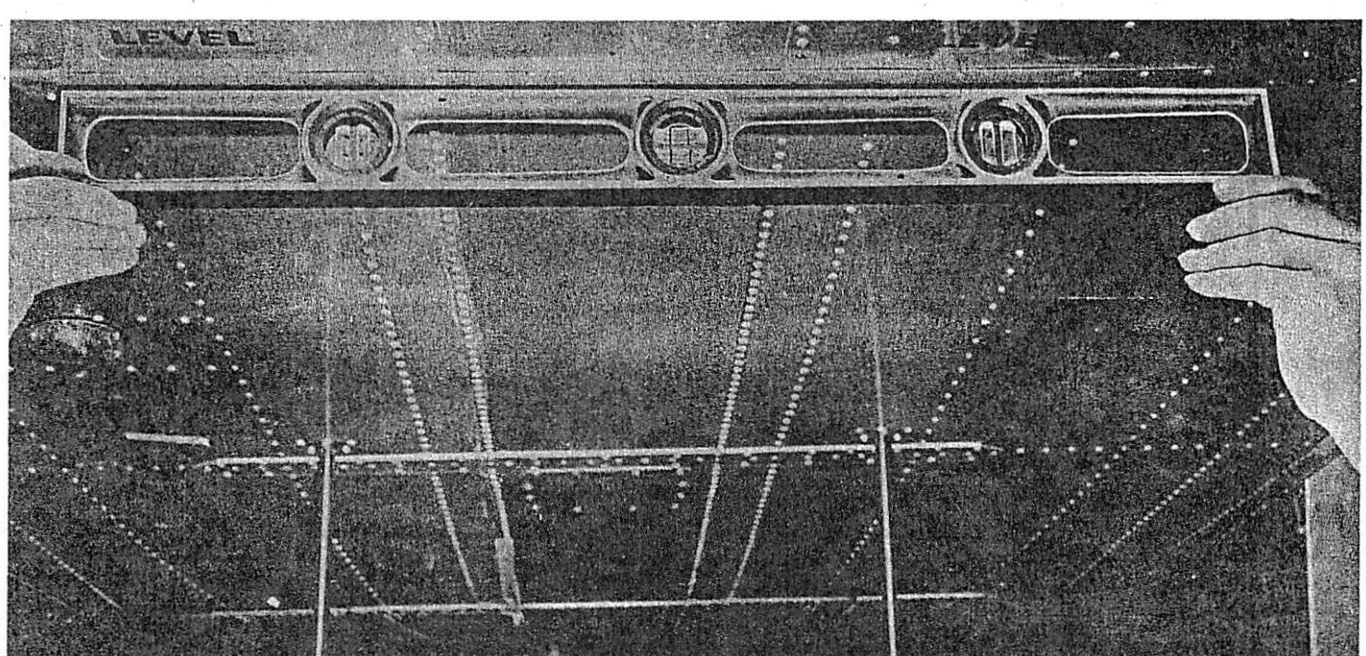


Figure 1-6. Lateral Leveling

1-25. CONTROL SURFACE REBALANCE DATA.

1-26. The control surfaces of this airplane were carefully balanced both dynamically and statically. Balance is obtained by the addition of lead ballast weight within the nose section of each surface.

1-27. BALANCING AILERONS.

1-28. The minimum overbalance of the left aileron is 1.6 inch pounds + 4 inch pounds -0 inch pounds. The minimum overbalance of the right aileron is 1.9 inch pounds + 4 inch pounds -0 inch pounds.

1-29. To balance an aileron, it is necessary to place it on a jig similar to the one shown in figure 1-7. Place a small platform scale under the leading edge and support it with a spindle. The chord line of the aileron must be horizontal at time of weighing. To determine the amount of static overbalance use the equation given below:

Static Overbalance = D (W-S)

D = Distance forward of hinge line that weight reading was taken, in inches.
W = Scale reading on platform scale, in pounds.
S = Weight of spindle used to support leading edge of surface on the scales, in pounds.

1-30. If the value obtained from the preceding equation is smaller than 1.6 inch-pounds on the left aileron

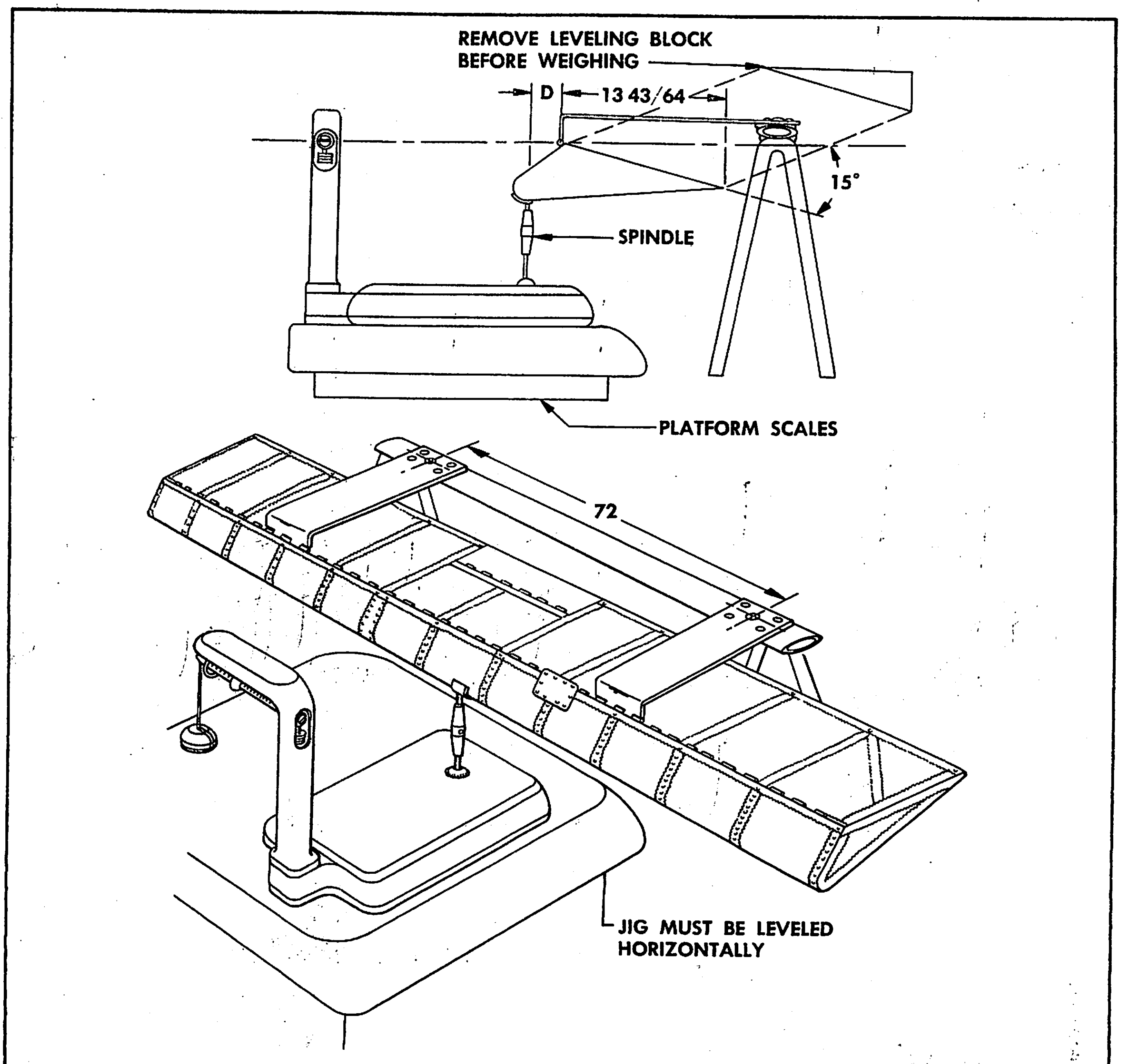


Figure 1-7. Balancing Ailerons

and 1.9 inch-pounds on the right aileron, it will be necessary to add balance weights to the aileron. Add lead washers (Beech Part 183809) inside of the nose of the inboard rib until proper balance is obtained. A maximum of 7 washers may be used. Washers may be attached to the rib by an AN526-10-18 screw, AN365-1032 nut, and two AN970-3 washers. Lead washers may be fabricated locally, if manufactured parts are not available, from 1/16-inch sheet lead. Make in form of a flat washers 1-1/2 inches in diameter with a No. 12 hole drilled in the center. If the value outlined in paragraph 1-28 cannot be obtained by the addition of a maximum of 7 washers (Beech Part 183809), use a section of bar weight (Beech Part 181308), until proper balance is obtained. When bar is used, attach inside of the inboard leading edge with a minimum of two AN 520-10-14 screws, two AN 365-1032 nuts, and two AN 970-3 washers. Hole spacing is not to exceed 3/4 inch and edge distance is not to exceed 1/2 inch. If it is necessary to remove any appreciable amount of lead (1/2 weight or more), all methods and equipment should be thoroughly checked to be sure the results are being interpreted correctly.

1-31. BALANCING RUDDERS.

1-32. The maximum static underbalance of the left rudder must not exceed 1.4 inch-pounds, while the static underbalance of the right rudder must not exceed .3 inch-pounds.

1-33. To balance the rudders, set-up jig similar to that shown in figure 1-8. Weigh on small platform scale, as described for the ailerons and use the following equation to determine the underbalance. Chord line must be horizontal at time of weight reading.

Static Underbalance=D(W-S)

D= Distance aft of hinge line that weight reading was taken in inches.
W= Scale reading on platform scale in pounds.
S= Weight of spindle used to support trailing edge of surface on the scale in pounds.

1-34. If the values obtained from the preceding equation are greater than 1.4 inch-pounds on the left rudder, and .3 inch-pounds on the right rudder, the situation

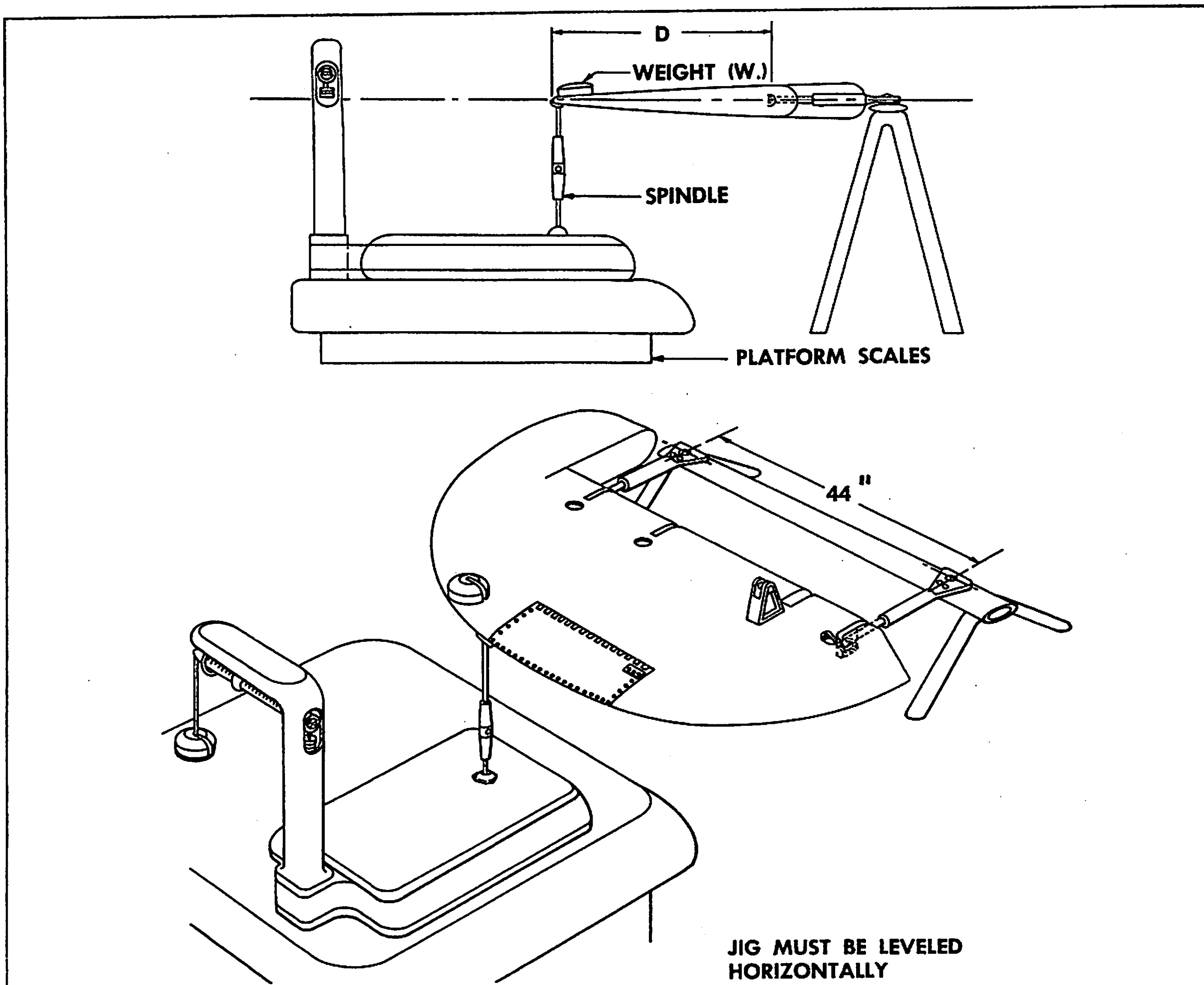


Figure 1-8. Balancing Rudders

must be corrected as follows: add washers (Beech Part 183809) inside the balance rib, Part 186253-4 (See figure 1-9) until correct balance is obtained. A maximum of seven washers, secured with an AN526-10-18 screw, AN970-3 washer and AN365-1032 nut is allowed.

1-35. BALANCING THE ELEVATOR.

1-36. The elevator is balanced dynamically by lead weights installed inside the leading edge. Required static balance is obtained when the center of gravity of the complete elevator is not more than three inches aft of the hinge center line. To balance the elevator it is necessary to place it in a jig similar to that shown in figure 1-8. Use the following equation to determine the center of gravity:

Dynamic Balance

$$CG = \frac{(Ws)\ (X)}{W}$$

The CG must be less than 3 inches.
Ws = Scale reading minus weight of spindle.
X = Distance from hinge line to spindle.
W = Total weight of complete elevator.

1-37. If the value obtained from the preceding equation is greater than 3 inches, the situation must be corrected as follows: ream a one-inch hole in either outboard rib. Secure a lead balance weight (Beech Part 196132). (This weight will have to be cut down to fit the contour of the leading edge). Drill a No. 10 hole through the leading edge of the elevator and the formed lead weight. Countersink the hole and install an AN510-10-24 screw, an AN970-3 washer, and an AN365-1032 nut. To determine the amount of lead weight needed, use the following equation:

Balance Equation

$$W1 = \frac{Ws\ (X) - 3W}{3 + L}$$

W1 = Minimum weight of lead to be added to obtain static balance.
Ws = Scale reading minus weight of spindle.
W = Total weight of complete elevator.
L = Distance from most forward portion of leading edge to hinge center line.

1-38. REBALANCING CONTROL SURFACES BY COMPUTATION.

1-39. The following method should be followed to determine the amount of weight necessary to restore correct balance requirements if facilities are not at hand to balance the surfaces by actual test.

a. Accurately determine the weight of the part or parts removed, if any. Measure the distance from the hinge centerline and note the location of the approximate center of gravity of the parts removed.

b. Accurately weigh the replacement parts and measure their location with reference to the hinge centerline.

c. With the data from the above procedure, substitute the proper values in Equation 1, step e, to determine the necessary amount of balance weight which must be added or removed to properly rebalance the surface.

d. The additional lead ballast weight must be located inside the leading edge of the surface as far forward as possible and at least as far outboard as the repair.

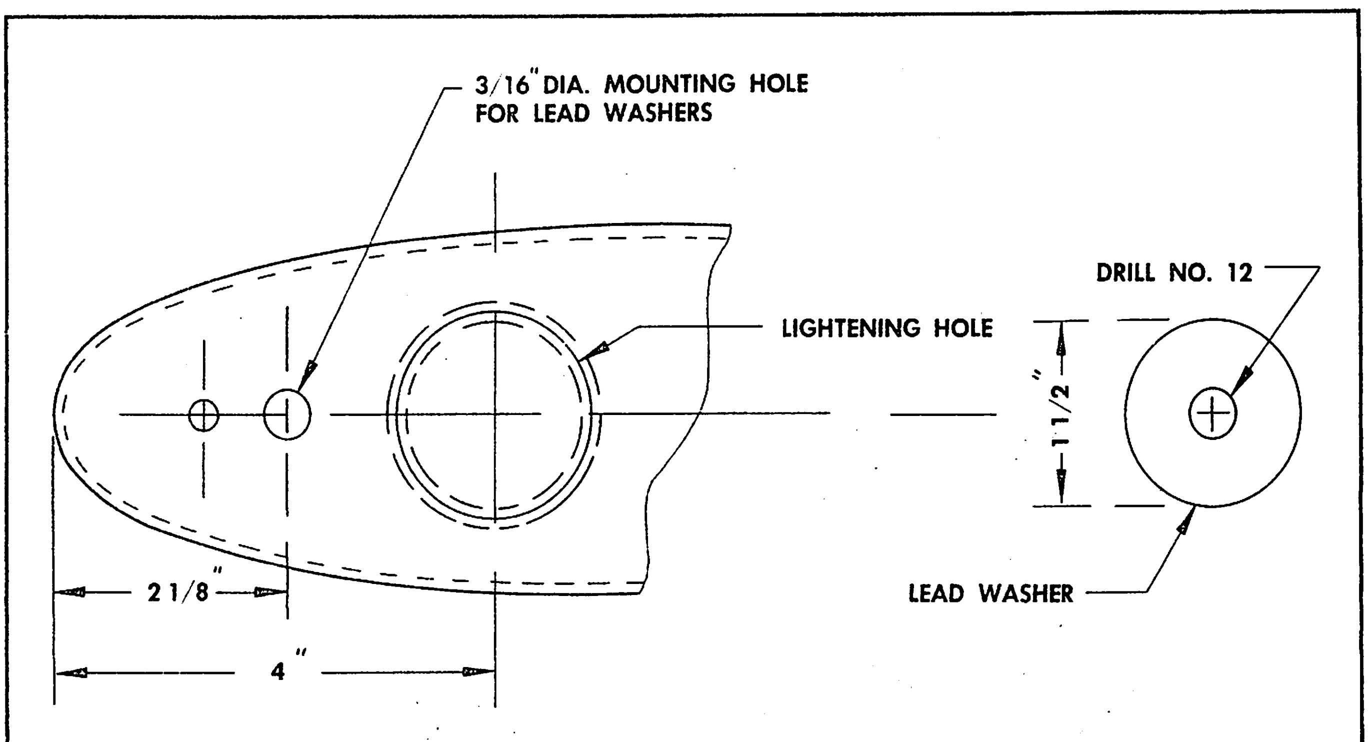


Figure 1-9. Installation of Lead Balance Weight in Rudder

e. Equation 1:

$$B = \frac{Wdr}{d1}$$

B = Ballast necessary at chosen location.
W = Weight increase due to repair.
dr = Distance of repair aft of hinge centerline.
d1 = Distance of ballast lead forward of hinge centerline.

f. The flaps are not dynamically balanced and need not be rebalanced after repairs.

1-40. FORMING SHEET STOCK.

1-41. Since many of the repairs described in this handbook involve forming of sheet aluminum stock, a short description of the methods used in forming sheet stock will be included here. Patches should be of the same type metal and the next gauge heavier. In order to form the heat-treatable alloys, it will be necessary to form them in the annealed or "O" condition, and heat-treat them to the "T" condition after forming and before the patch is installed on the aircraft.

a. High strength rolled sheet is used for stressed skin and corrugations; formed sheet for the bulkheads and ribs. The material in all replacements or reinforcements will be chosen on the same basis as the material in the original structure. If it is necessary to substitute an alloy weaker than the original, a larger gauge must be used to give equivalent cross-section strength; however, the reverse is not a safe practice. Never use a smaller cross section, even when using a stronger metal. All contacting surfaces, regardless of the material composition, must be prepared and all corrosion removed before permanent installation.

b. Particular care will be taken in selecting material when forming is required. The heat-treated and cold-worked alloys will stand very little bending without cracking. The soft alloys are easily formed, but are not strong enough for primary structures. The strong alloys can be formed in their annealed or "O" condition, and heat-treated to develop their strength before assembling. In some cases, where the annealed metal is not available, it is possible to heat the metal, quench it according to regular heat-treating, and then form it before the age hardening sets in. The forming must be completed in a half-hour after quenching or the material will become too hard to work. If a brake is used to form sections, it is necessary that a thin piece of soft metal be used over the brake jaws to prevent scraping and scratching and damage to the surface of the sheet. It has been found that cold-rolling and forming operations which can be made on steel without injuring the sheet will sometimes, in the case of aluminum alloy sheet, cause the metal to flow away from the point of maximum pressure resulting in a sheet which is thinner at certain points.

NOTE

Bend lines must be drawn in pencil on aluminum alloy and not scratched in. Wherever possible the bends should be across the grain of the metal.

1-42. RIVETS AND RIVETING.

1-43. STANDARD SOLID SHANK RIVETS. Standard solid shank rivets are generally used in aircraft construction. They are fabricated in the following head styles: roundhead, flathead, countersunk head, and brazier head. Roundhead rivets are generally used in the interior of the aircraft except where clearance is required for adjacent members. Flathead rivets are generally used in the interior of the aircraft where head clearance is required. Countersunk head rivets are used on the exterior surfaces of the aircraft to minimize turbulent air flow. Brazier head rivets are used on the exterior surfaces of the aircraft where strength requirements necessitate a stronger rivet head than that of a countersunk head rivet. Both the brazier head and countersunk head rivets are used in the interior of the aircraft where head clearance is required.

1-44. STANDARD AN RIVETS. Standard AN rivets are identified in figure 1-10. The 2S rivet is made of pure aluminum and is used only in riveting nonstructural parts fabricated from the softer aluminum alloys, such as 2S (any temper) and 52SO. The A17ST rivet is in general use throughout aircraft structures for riveting parts fabricated from the heat-treatable aluminum alloys. The 24ST rivets are used where their higher strength properties are more desirable than those of the A17ST rivet. See Tables I and II.

1-45. CHERRY BLIND RIVET. Cherry rivets are patented blind rivets used where strength requirements permit, when one side of the structure is inaccessible, making it impossible or impractical to drive standard solid shank rivets. Cherry rivets have higher defection rates in shear than standard solid rivets. For this reason, it is not advisable to replace any considerable number of solid rivets in a given joint by Cherry rivets, inasmuch as this may result in overstressing the remaining solid rivets. The following specific instructions apply:

a. The hollow Cherry rivet IS-1112 and IS-1113 shall not be used.

b. The Cherry rivet used shall be of the same or greater strength and one size larger than the rivet it replaced. Cherry rivets may be replaced size for size.

c. When possible, the exposed end of each clipped plug shall be coated with a 10-percent chromic-acid solution or with zinc chromate primer.

d. If blind fasteners other than Cherry rivets are encountered, it is recommended that replacements be made by either a Cherry or standard rivet.

1-46. STANDARD HOLES FOR RIVETS. The following table specifies the size hole to drill for the application of various sizes of rivets:

Rivet Diameter	Drill Size
1-16	No. 50
3/32	No. 41
1/8	No. 30
5/32	No. 21
3/16	No. 11
1/4	No. F
5/16	No. P
3/8	No. W

1-47. RIVET REQUIREMENTS.

1-48. To compute the number of rivets to be used in a patch, use the following procedure:

a. First, it is necessary to compute the design load. For example, assume a crack 2-1/2 inches long in a sheet of 24ST alclad aluminum which is .032 inch thick. 24ST alclad sheet has an ultimate tensile strength of 60,000 pounds per square inch. See Table V. Estimate the design load as follows:

Design Load = Length of crack x L x gauge of skin.
L being allowable ultimate tensile strength.
Thus, Design Load = 2.5 x 60,000 x .032 = 4800

b. It is possible to use rivets of several different sizes in making a repair. Good design practice is to make a joint which has approximately equal strength in shear and bearing, with the gearing strength slightly higher, if possible. A general rule for selecting rivets of the proper diameter to join aluminum-alloy sheets is to select a size having a diameter approximately three times the thickness of the heavier sheet in the patch. An A17ST rivet with a diameter of 1/8 inch has been selected for this example. The allowable shear and combinations of rivet diameter, sheet material, and gauge thickness are given in the following tables. The permissible bearing load per rivet must be based on the thinner material where different gauges of material are used in the repair. The allowable loads for one A17ST rivet 1/8 inch in diameter, as given in the tables are: allowable shear load - 374 pounds; allowable bearing load - 364 pounds. The smaller of these values must be used as the allowable load per rivet; in this example, 364 pounds.

c. The ratio between the diameter of the rivet hole and the distance from the hole to the edge of the material is a critical factor in the bearing strength of rivets. In the tables, ratios of 1.5 and 2.0 have been used; i.e., in Table III, the calculations were based on the edge distance being 1.5 times the hole diameter, and in Table IV, the edge distance is twice the hole diameter. In using the tables, these ratios must be observed; wherever possible, they should be maintained or exceeded in making repairs. The 1.5 ratio is generally accepted as a minimum in good aircraft practice.

d. The number of rivets required on each side of the crack may now be determined.

$$\text{Number of rivets required} = \frac{\text{Design Load}}{\text{Allowable Load per Rivet}}$$

$$\text{Number of rivets required} = \frac{4800}{364} = 13.2 \text{ (use 13)}$$

TABLE I

SHEAR STRENGTHS OF PROTRUDING HEAD A17S-T3 ALUMINUM RIVETS - SINGLE SHEAR

	RIVET DIAMETER							
Sheet Thickness*	1/16	3/32	1/8	5/32	3/16	1/4	5/16	3/8
.016	102							
.018	104							
.020	106							
.025		211						
.032		217	374					
.036			380					
.040			386	575				
.045			388	584				
.051				594	838			
.064				596	862	1494		
.072						1519	2371	
.081						1544	2396	
.091						1550	2421	
.102							2450	3412
.128							2460	3510
.156								
.188								
.250								

*Thickness of Thinnest Sheet

TABLE II

SHEAR STRENGTHS OF PROTRUDING HEAD A17S-T ALUMINUM RIVETS - DOUBLE SHEAR

	RIVET DIAMETER							
Sheet Thickness*	1/16	3/32	1/8	5/32	3/16	1/4	5/16	3/8
.016	146							
.018	160							
.020	168							
.025	184	310						
.032	198	355	534					
.036	206	372	574					
.040	209	389	615	820				
.045	212	400	645	882				
.051		417	675	944	1231			
.064		434	726	1053	1410	2133		
.072			756	1095	1477	2294		
.081			776	1130	1545	2455	3385	
.091				1161	1590	2576	3705	
.102				1192	1657	2697	3897	5012
.128					1724	2899	4344	5742
.156						3060	4600	6199
.188						3100	4792	6564
.250							4920	7020

*Sheet Thickness is That of Middle Sheet

TABLE III

BEARING STRENGTH OF RIVETS ON 24ST ALCLAD SHEET HEAT TREATED BY USER

	RIVET DIAMETER							
Sheet Thickness E/D=1.5	1/16	3/32	1/8	5/32	3/16	1/4	5/16	3/8
.016	83							
.018	94	135						
.020	105	150						
.025	131	187	250					
.032	167	239	321	397				
.036	188	270	361	446	537			
.040	209	300	401	496	596			
.045	236	337	451	558	671			
.051	267	382	511	633	760	1022		
.064	341	488	653	811	970	1304	1646	
.072	383	549	735	906	1089	1471	1852	2210
.081	432	619	827	1026	1232	1654	2083	2488
.091	485	695	930	1153	1383	1860	2337	2790
.102	543	778	1041	1288	1518	2083	2616	3132
.128	682	978	1304	1614	1916	2616	3283	3927
.156	835	1193	1598	1972	2369	3188	4007	4794
.188	994	1431	1916	2369	2846	3832	4818	5756
.258	1328	1908	2552	3156	3792	5104	6416	

E=Edge Distance
D=Hole Diameter

TABLE IV

BEARING STRENGTH OF RIVETS ON 24ST ALCLAD SHEET HEAT TREATED BY USER

Sheet Thickness

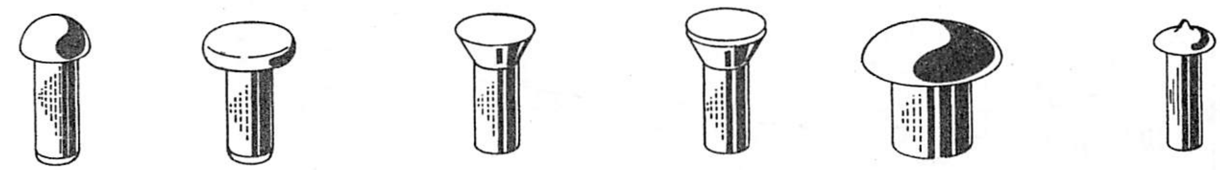
	RIVET DIAMETER							
E/D=2.0	1/16	3/32	1/8	5/32	3/16	1/4	5/16	3/8
.016	95							
.018	107	153						
.020	119	170						
.025	149	212	284					
.032	189	272	364	450				
.036	213	306	410	506	609			
.040	237	340	455	563	676			
.045	267	382	512	634	761			
.051	303	434	580	718	862	1159		
.064	393	562	752	933	1116	1501	1894	
.072	441	632	846	1043	1254	1693	2132	2544
.081	497	712	952	1180	1418	1903	2397	2864
.091	558	800	1071	1327	1592	2141	2690	3212
.102	625	896	1199	1482	1748	2397	3010	3605
.128	785	1125	1501	1857	2205	3010	3779	4520
.156	961	1373	1839	2269	2727	3699	4612	5517
.188	1144	1647	2205	2727	3276	4410	5545	6625
.250	1528	2196	2937	3633	4365	5874	7384	

E=Edge Distance
D=Hole Diameter

TABLE V

ULTIMATE TENSILE STRENGTH OF ALUMINUM SHEET

Type of Material	Thickness	Tensile Strength (psi)
24ST Alclad Sheet	Less than .064	60,000
24ST Alclad Sheet	.064 or greater	63,000
24ST Aluminum Alloy Sheet	Less than .5	65,000
61S-T6 Aluminum Alloy Sheet	Less than .25	42,000
52S-1/2H Aluminum Alloy Sheet	Less than .25	34,000
52SH Aluminum Alloy Sheet	Less than .25	39,000



AN 430	AN 422	AN 425	AN 426	AN 455 AN 456	AN 470
ROUND HEAD	FLAT HEAD	COUNTER SUNK HEAD	COUNTER SUNK HEAD	BRAZIER HEAD	UNIVERSAL HEAD

HEAD MARKINGS	MATERIAL	AN STD. RIVET MATERIAL							AN MATERIAL CODE	CONDITION	HEAT TREAT BEFORE USING
		425	426	430	442	455	456	470			
PLAIN	2S	X		X	X			X	A	AS FABRICATED	NO
RAISED CROSS	56S		X	X				X	B	AS FABRICATED	NO
DIMPLED	A17ST	X	X	X	X	X	X	X	AD	(T) HEAT TREATED	NO
RAISED DOUBLE DASH	24 ST	X	X	X	X	X		X	DD	(T) HEAT TREATED	YES

RIVET NUMBER CODE
AN 430-AD5-12

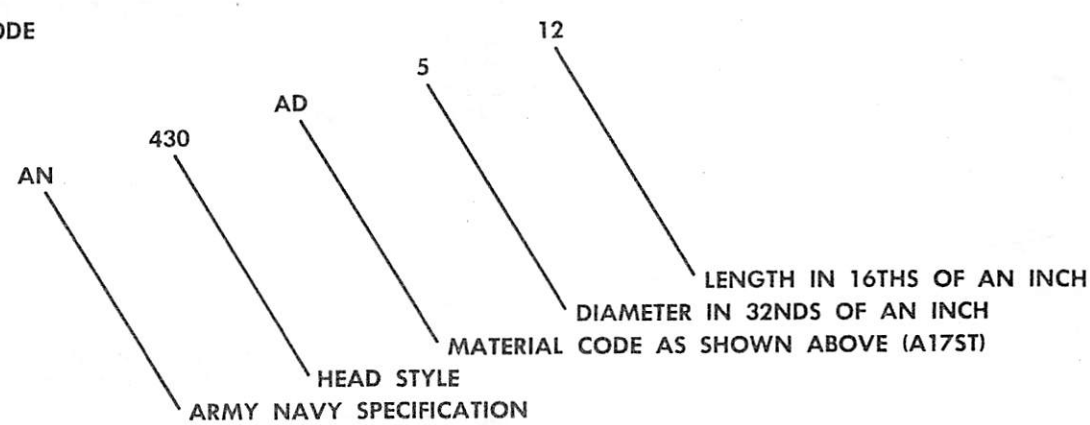


Figure 1-10. Rivet Identification

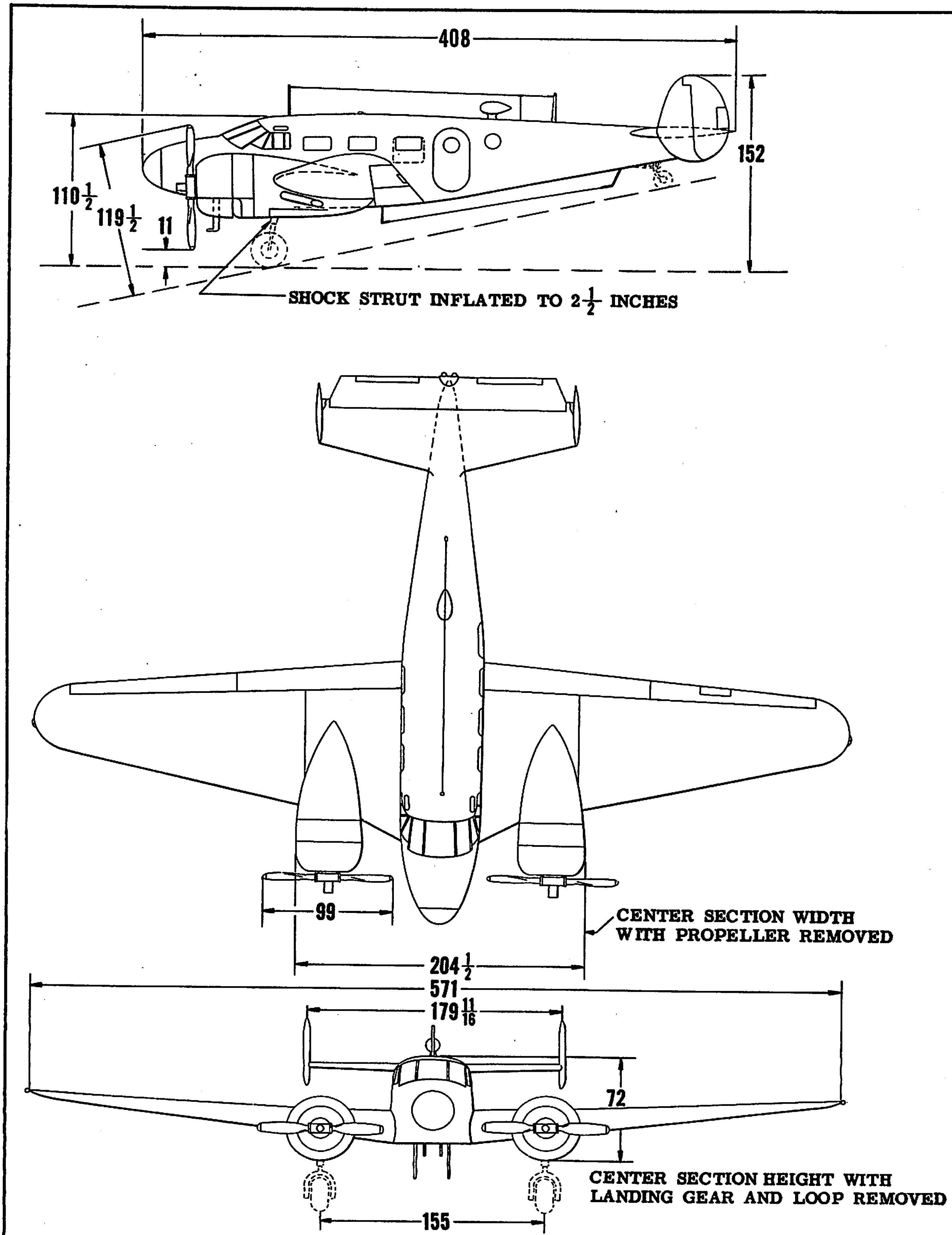
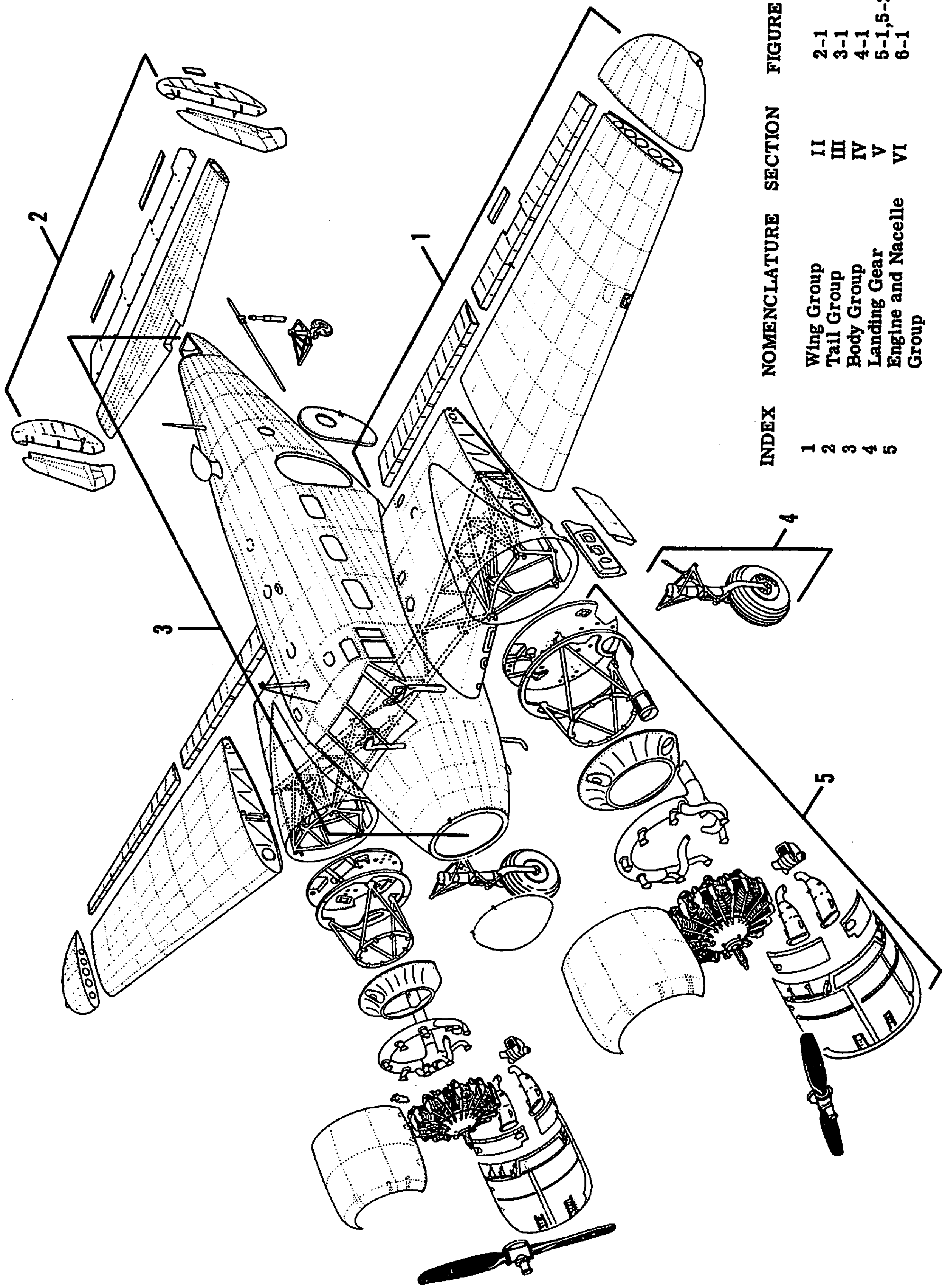


Figure 1-11. Rigging Dimensions



INDEX	NOMENCLATURE	SECTION	FIGURE
1	Wing Group	II	2-1
2	Tail Group	III	3-1
3	Body Group	IV	4-1
4	Landing Gear	V	5-1,5-2
5	Engine and Nacelle Group	VI	6-1

Figure 1-12. Exploded View of Aircraft

SECTION II

WING GROUP

TABLE OF CONTENTS

2-1. WING GROUP.

2-2. DESCRIPTION.

2-3. The wing is an all metal, cantilever type composed of three sections: center section wing, outboard wing, and wing tip. The spar of the outboard wing attaches to the center-section truss at upper and lower spar fittings with specially tapered spar bolts and nuts. A special nut and bolt also secures the outboard wing rear spar to the rear spar of the center section wing stub. The wing tip is of the same general construction as the outboard wing. It is attached by a hinge wire through the front spar and by machine screws through the skin of the wing panel and wing tip. See figure 2-1.

2-4. CENTER SECTION WING.

2-5. DESCRIPTION.

2-6. The primary structure of the center section consists of a single, welded, heat treated, tubular steel truss. This truss carries the fittings for the engine mounts, landing gear, and outer wing panel main spar. Recesses are provided in the center section for the installation of four fuel tanks: one 78 U. S. gallon and one 25 U. S. gallon on each side. Also provided are recesses to house two batteries, one on each side. The remainder of the center section structure consists of aluminum-alloy ribs, bulkheads, stringers, and a smooth skin covering. The rear spar is a shear beam which supports the center and inboard flap hinges. Removable panels are provided over the fuel tanks and battery box openings. See figure 2-2.

2-7. OUTBOARD WING.

2-8. DESCRIPTION.

2-9. The outboard wing panels are constructed with single, steel tube front spars and aluminum-alloy skin. Rear spars transfer the shear load between upper and lower skin. The remainder of the structure consists of ribs and stringers covered with alclad skin (figure 2-3).

2-10. WING TIP.

2-11. DESCRIPTION.

2-12. The wing tip consists of a single lateral rib which parallels the outer panel ribs, a wing tip bow, skin, and intersecting bulkheads (longitudinal ribs).

2-13. AILERON.

2-14. DESCRIPTION.

2-15. The fabric-covered aileron is constructed of aluminum alloy main and nose ribs attached to a spar. A metal nose plate forms the leading edge, and the trailing edge is made of standard Alcoa K-1508 rolled section. The aileron is attached to the outboard wing by a piano-type hinge extending the full length of the aileron.

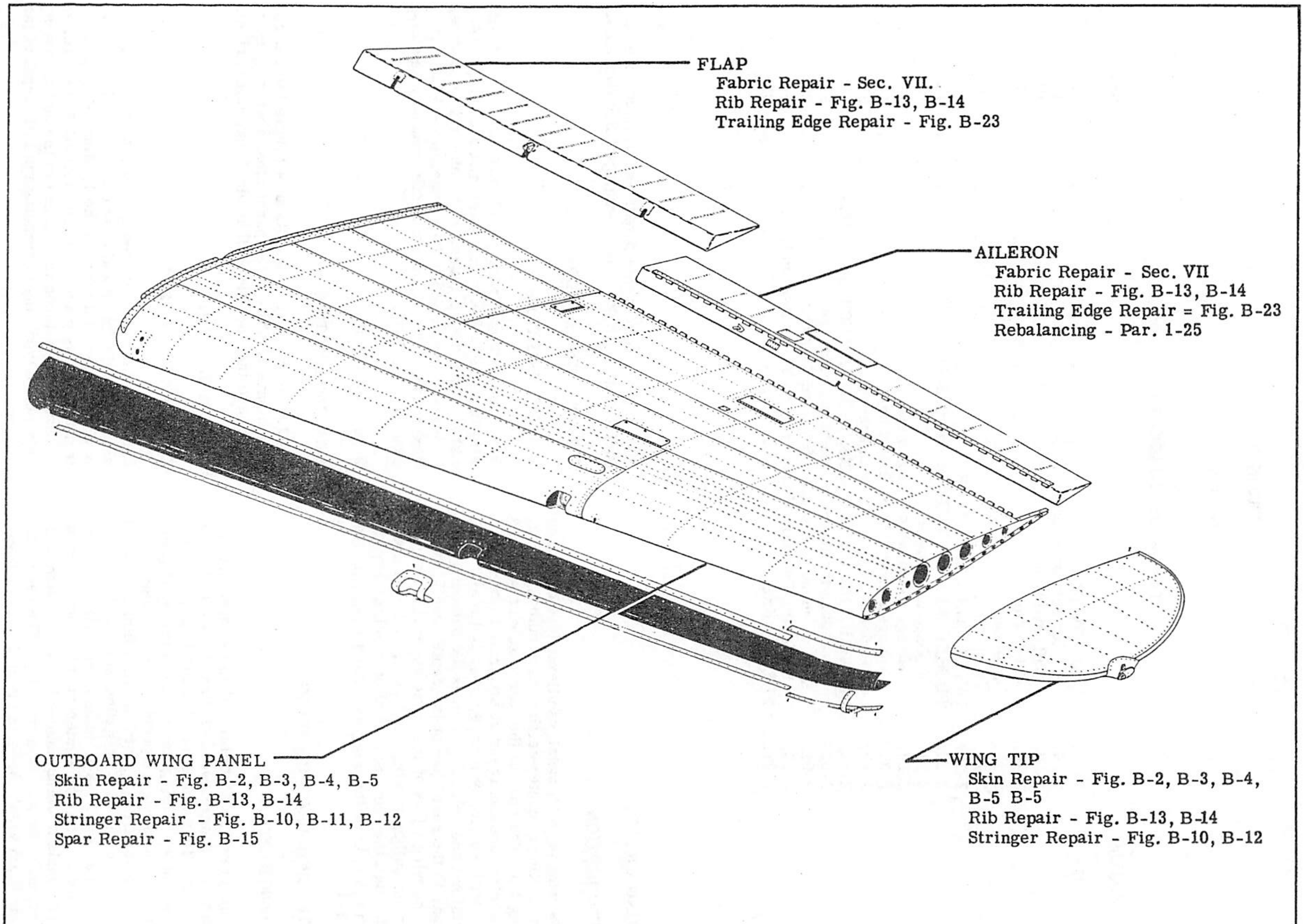


Figure 2-1. Wing Group Exploded

2-16. FLAP.

2-17. DESCRIPTION.

2-18. The construction of the fabric covered flap is fundamentally the same as that of the aileron, figure 2-1. The flap is hinged to three brackets on center section wing and inboard part of the outboard wing. The undersurface is protected from exhaust and wheel blast by a metal shield attached to the flap rib by rivnuts.

2-19. DEFINITION OF DAMAGE.

a. Negligible damage is damage that will not affect the airworthiness of the wing assembly and does not require immediate attention.

b. Damage repairable by patching is damage that may be repaired by covering or reinforcing a portion of the wing skin or structure.

c. Damage repairable by insertion is damage that requires the removal and replacement of a portion of the wing skin or structure.

d. Damage necessitating replacement of parts is damage unrepairable by patching or insertion, but which may be repaired by installing a new part. Damage requiring replacement, but which cannot be replaced because of structural design, will necessitate replacement of the entire assembly.

2-20. PROCEDURE FOR REPAIR OF DAMAGE.

2-21. NEGLIGIBLE DAMAGE.

2-22. Damage of this classification shall be limited for the entire wing assembly to surface dents in the skin. These dents should not substantially change the contour of the airfoil, and must be carefully investigated for indications of structural damage.

2-23. DAMAGE REPAIRABLE BY PATCHING.

2-24. Skin - Small holes, cracks, or breaks in the metal skin covering of the wing assembly may be patched in different ways. It is advisable to use a flush-type patch (See figure B-3 and B-5) in patching the leading edge of wing. This section, that part forward of the front spar, must be maintained in smooth contour so as not to interrupt the flow of air. Other sections of the wing may be repaired with surface patches of the following types:

a. Round holes up to one inch in diameter may be reinforced by a washer installed as shown in figure B-4. Reinforcing the hole rather than covering it, permits the use of a bucking bar in setting the rivets. If Cherry rivets are used, a disc may be substituted for the washer.

b. Small breaks or punctures in the skin may be repaired as shown in figures B-2 and B-3. The ragged edges should be cut away, as shown by the dotted lines, so that no sharp corners remain. To compute rivet size and spacing, refer to paragraph 1-48 and to figure 1-10.

NOTE

In the process of manufacturing, 24ST sheet is now used for skin plating. Repairs made to skin surfaces may be made with 24ST regardless of the type of material which is being replaced.

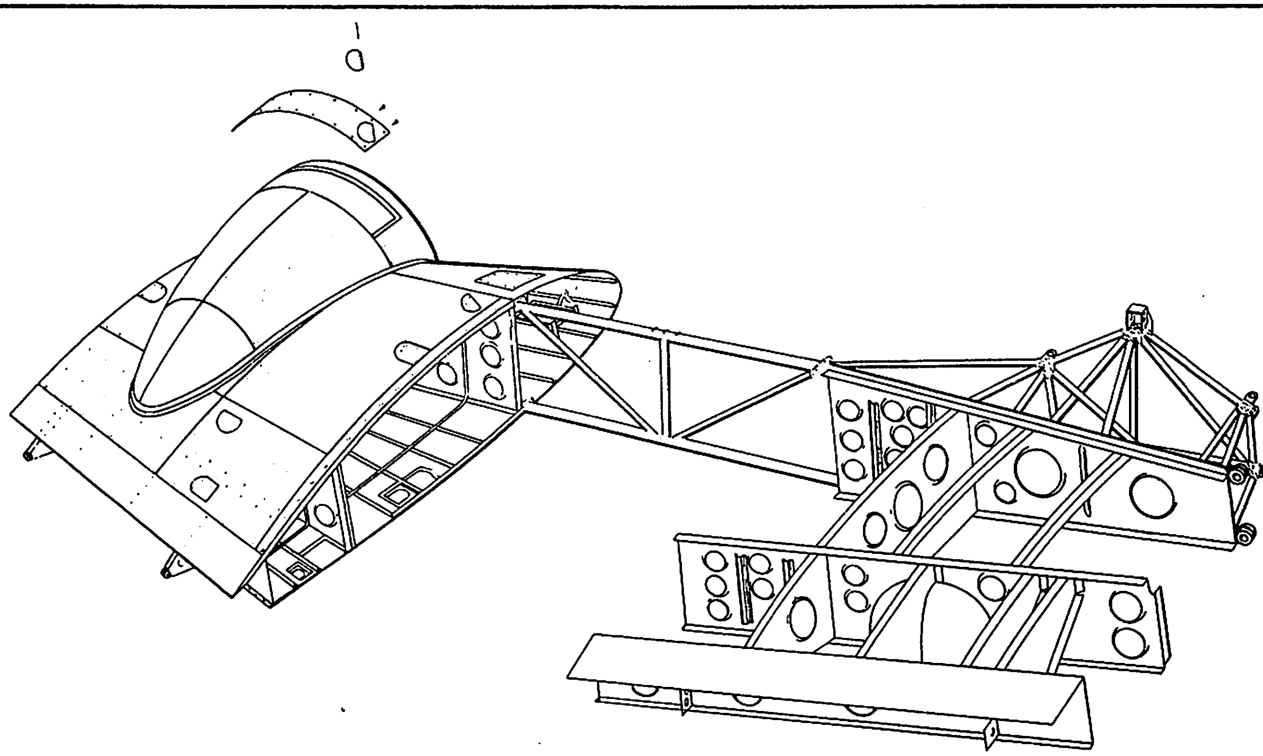


Figure 2-2. Center Section Wing

c. To repair cracks in the skin, first stop drill holes at each end or sharp corner of the crack (See figure B-1). The patch plate should be cut large enough to clear all parts of the crack by 7/8-inch. Cherry rivets, if available, may be substituted for the standard rivets.

2-25. STRINGERS.

2-26. Cracks in the stringers of the wing assembly may be repaired by reinforcing the damaged member with an extruded section or a hand-formed splice angle as follows:

a. An extrusion of the same cross section as the original stringer may be used as patch material. The reinforcement must fit smoothly into the stringer; if necessary, the bulb angle and corner radius of the patch extrusion may be filed. (See figure B-10). It is also permissible to attach the reinforcement on the opposite side of the stringer.

b. The hand-formed splice angle should be made of .064, 24ST sheet and installed as shown in figure B-11. Radius of bend for this angle should be 3/16-inch.

c. If it is necessary to splice more than one stringer, splices should be staggered in such a manner that no two splices fall at the same station. (See figure B-12).

2-27. RIBS.

2-28. Rib structure of all components of the wing assembly is essentially the same, and repair methods will apply to all ribs. Repair as follows:

a. Small cracks or breaks, entirely within the rib web and where space allows, may be repaired by methods described for skin patches in paragraph 2-24.

b. Damage in formed flanges, in the flange and extending down into the web, or in ribs too small for a skin type patch will require a patch formed to the flange of the rib (See figure B-13 and B-14). The crack or break must be covered with a patch of at least 3/4-inch edge margin. Stop holes should be drilled at the end of cracks, and breaks should be properly trimmed to prevent spreading. In riveting wing rib patch to rib flange, skin rivets may be used with an additional rivet between the original rivet holes. Rivets must not be spaced closer than 3/8-inch.

2-29. SPARS.

2-30. Wing rear spar sections and flight control surface spars may be patched as illustrated in figure B-15. All cracks or breaks in spar members shall be rein-

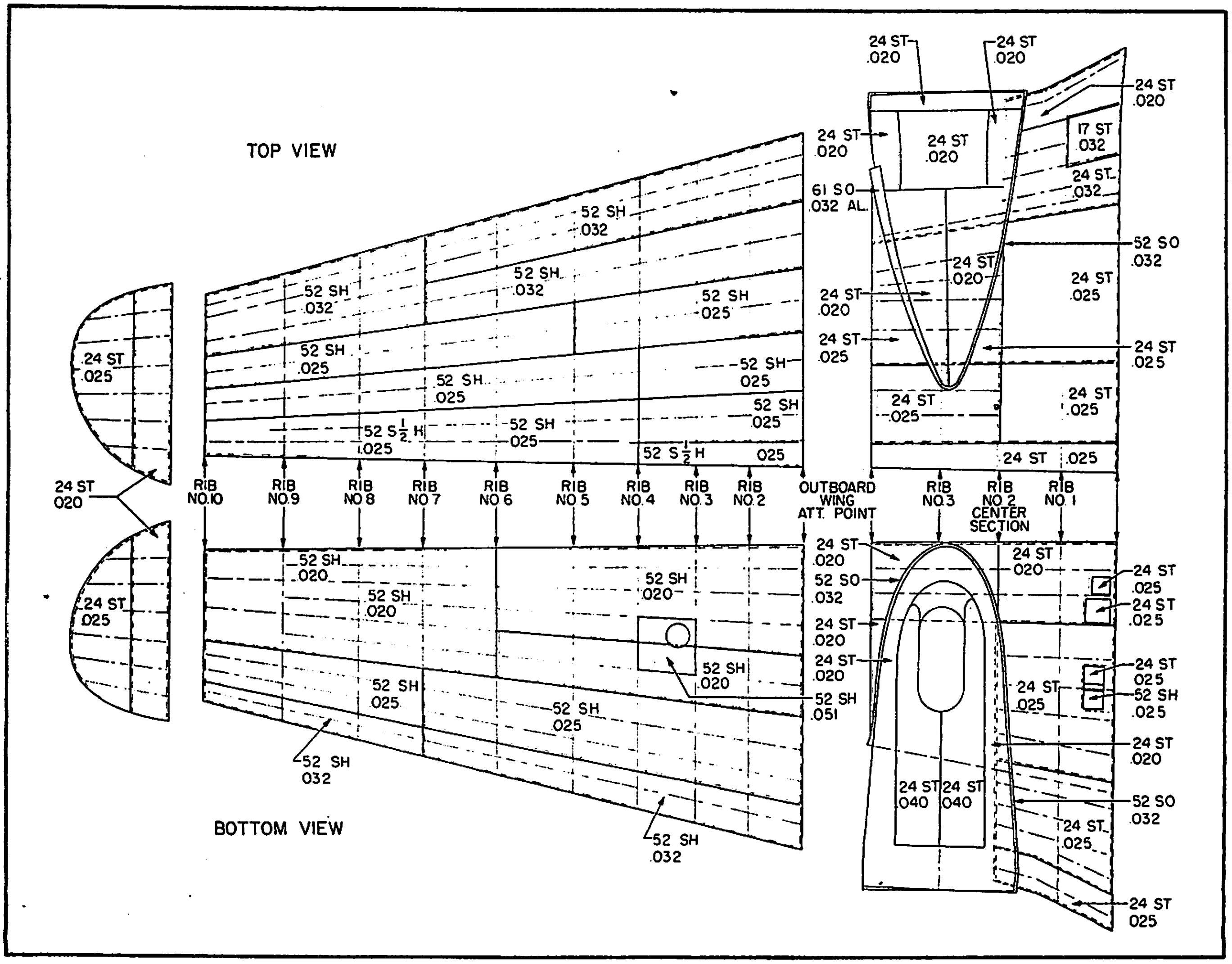


Figure 2-3. Wing Skin Plating Diagram

forced with a patch plate extending across the full cross section of the spar. A patch on the wing rear spar may be riveted to the wing skin, by enlarging the old rivet holes and using the next larger size rivet, or spacing new rivets midway between the old rivet holes.

2-31. BULKHEADS.

2-32. Bulkheads in the center section and wing tip are suited to repair methods similar to those for wing skin patching. (Refer to paragraph 2-24).

2-33. TRAILING EDGE REPAIRS.

2-34. The trailing edges of the aileron and flap form the wing assembly trailing edge. Cracks and small breaks between rib members may be repaired by a reinforcing patch as shown in figure B-23. Care must be exercised in inserting the splice block, to avoid further damage to the edge member. A minimum of three rivets must be used on each side of the break.

2-35. SUPPORTING STRUCTURE.

2-36. Due to the size and type of construction of supporting members such as flap hinges and flight control pulley brackets, damage of these parts will necessitate replacement. Other supporting structures, such as battery support and ribs, can be repaired. The battery support structure consists of a frame of channel sections, riveted to the fuselage at one end to Rib 1 of the center section wing at the other. Cracks or breaks in the channel sections may be patched as shown in figures B-18 and B-3.

2-37. DAMAGE REPAIRABLE BY INSERTION.

2-38. SKIN.

2-39. When damage to the skin of the wing assembly is too extensive to be repaired by a reinforcing patch, the skin should be cut back to a supporting structure such as a rib or stringer. A new section of skin should be fitted into place and riveted to the structure. When determining the amount of skin to remove, avoid making a new joint on a structural member that will be a continuation of an original skin joint. It is undesirable also to make a skin joint on both the upper and lower surfaces on the same rib or stringer. It is better to extend the replacement skin to the next member rather than to make such joints as described. The replacement skin should be of the next greater thickness than the damaged skin and of 24ST material.

2-40. Skin joints are classified as either longitudinal or transverse:

a. The longitudinal skin joints lie along stringers or spars. If the new joint is made along a member where there was an old skin joint, the same rivet spacing may be employed. If there was no joint previously, the rivets should be spaced only half as far apart as the old rivets, as shown in figure B-9.

b. The transverse skin joints lie along the ribs and the skin is attached as shown in figure B-8. The original rivet spacing may be used if the joint is made at the previous joint, but if there has been no previous joint, the replacement skin should be cut to lap over the rib by 1/2-inch. It should be riveted with a staggered row of rivets with spacing equal to the original. The 1/2-inch lap may be made on either side of the rib.

2-41. STRINGERS.

2-42. Damage requiring removal of a portion of the stringer may be repaired by cutting out the damaged portion and inserting a new extruded section. This section may be spliced into place by methods described in paragraph 2-26.

2-43. RIBS.

2-44. Serious damage to a rib member may require replacement of a portion of the rib. The damaged part of the rib should be removed by making a straight cut across the rib. A new portion may then be made, using material of equal or greater strength and thickness. This new part should be spliced to the rib by using a splice plate of the same material as the rib and at least two inches wide. There should be a double row of 1/8-inch rivets on either side of the joint. The splice plate should be formed to extend inside of the rib flange and to rivet through the flange to the skin (see figure B-14).

2-45. SPARS.

2-46. Damage to the wing rear spar, and the aileron or flap spars, may be repaired by cutting away the damaged portion and forming a new section to replace it. The new section may be spliced to the original spar as shown in figure B-15. Rivet the wing skin to the spar by either drilling a size larger hole through the old rivet holes of the skin and using a size larger rivet, or space new rivets midway between the old rivet holes.

2-47. TRAILING EDGE REPAIRS.

2-48. To repair a damaged trailing edge of the flap or aileron, cut out the damaged section and replace it with a new piece. The splice should be made as shown in figure B-23. It is usually most convenient to make the extremities of the cut center between the trailing edge ribs. This permits the splice block (for attaching the new section) to be installed more easily. The splice block may be of 52SH or any stronger aluminum alloy. A minimum of three rivets on each side of the cut will be necessary.

2-49. DAMAGE REPAIRABLE BY REPLACEMENT.

2-50. SKIN.

2-51. If more then 50 percent of a skin panel is damaged, it is usually advisable to replace the complete panel. Use material of the next greater thickness and the same alloy strength and install as described in paragraph 2-39.

2-52. STRINGERS.

2-53. It is possible to replace the stringers in the wing assembly, but because of the amount of skin that would have to be removed, it is not practical. Make all repairs either by patching or insertion.

2-54. RIBS.

2-55. Replace ribs as follows:
a. Wing Ribs - Because of the integral construction of the ribs with the front spar, it is impractical to put in an entire new rib. However, a new nose section can be installed by splicing to the rib just forward of the front spar, and similarly, an aft rib section can be replaced by splicing into the rib immediately behind the front spar. Use the procedure for repairing by insertion described in paragraph 2-44. This is a difficult process because so much skin must be removed, but it can be accomplished satisfactorily.
b. Aileron and Flap Ribs - Extensive damage to these members is cause for replacement. Nose and tail sections of the rib are separated by the spar and are replaced as individual parts. If new parts are not available, replacement can be hand-formed.

2-56. SPARS.

2-57. FRONT SPAR.

2-57A. The wing front spar is a heat-treated member and is constructed integrally with the wing ribs; therefore, its removal and reinstallation is a complex and critical procedure. Detailed instructions are given in paragraphs 2-57F and 2-57G and in the six sheets of figure 2-3A. When attempting this repair, these instructions must be followed exactly.

2-57B. REAR SPAR.

2-57C. The outer panel rear spar consists of two assemblies spliced together at Rib 4. It is joined to the center section rear spar by a special bolt. Replacement of these assemblies is difficult, but practical.

2-57D. FLAP AND AILERON SPARS.

2-57E. Damage sufficient to warrant replacement of a flap or aileron spar will require replacement of the entire assembly.

2-57F. REPLACEMENT OF THE OUTER WING PANEL FRONT SPAR.

2-57G. The single outer wing panel front spar of heat-treated, welded steel tube is sufficiently stressed to withstand normal service loads. However, if the spar is damaged and it is known or suspected that the damage will reduce its strength, it should be replaced rather than attempting a temporary repair. The following procedure must be used:
a. Remove the outer wing panel and place it on a suitable fixture which will support it rigidly and at the same time provide a convenient working location.
b. With the wing properly supported, follow the detailed disassembly and reassembly procedures given in figure 2-3A, sheets 1 through 6.

CAUTION

Take care at all times that the aerodynamic design of the wing is not altered.

2-58. WING TIP.

2-59. Damage to the leading edge or extensive structural damage will necessitate replacement of the wing tip, due to difficulties of repair.

2-60. SUPPORTING STRUCTURES.

2-61. Due to the size and type of construction of supporting members such as flap hinges and flight control pulley brackets, damage incurred will necessitate replacement of the parts.

2-63. PREPARATION FOR REPAIR OF WELDED STRUCTURES.
a. Inspect the damaged area to ascertain that it does not extend beyond the limits illustrated in the repair diagram for the member in question.
b. Cut the tube to fit as shown in the illustration.
c. Obtain material for sleeves and cut to size as shown in the illustration. The closest possible fit must be maintained between the sleeve and the tube. If it is necessary to ream the sleeve, do not remove more than .010 inch of metal from the tubing wall.
d. Pack the area immediately adjacent to the repair with asbestos mud to minimize heat transfer.

WARNING

Drain and purge or remove all fuel tanks and remove the vent plugs from the tubes being welded. Failure to do so may result in an explosion.

2-63A. PROCEDURE FOR REPAIR OF WELDED STRUCTURES. The equipment recommended for weld repairs is the Lincoln Welder, Type SAE 300J which is rated at 30 volts, 3-phase, 60 cycle, 16 amperes with Lincontrol SL9053. With Lincontrol, use approximately 25 volts and Airco Rod 190. The Lincontrol is actuated by the operator's foot to regulate voltage and to eliminate globules at the end of the weld bead. Since the center section structure is a heat treated assembly, all welds must be of the highest possible quality and should be accomplished by certified welders who are thoroughly familiar with this type of weld repair. Make all welds in still air under temperature conditions which will not allow the welds to cool too rapidly. Magnaflux all welds; if possible, the repair weld should be magnafluxed the day following welding. Carefully examine the results of the magnaflux inspection, since major and minor defects will be revealed. In case of doubt, reject a weld rather than risk a failure in service.

2-63B. PROCEDURE FOR REPAIR OF CRACKED AREAS. Cracks in the center section truss fittings may be repaired only to the extent shown in figures 2-5, 2-6 and 2-7. Use the following procedure:
a. Grind out or vee cracks to a minimum of one half the thickness of the structure wall or web, using a 1/8-inch rotary file. Grind sufficiently to assure complete weld penetration.
b. Isolate the repair area by packing asbestos mud immediately adjacent to the crack so as not to preheat the adjoining areas.
c. Preheat with a No. 22 tip (Smith orifice), using a

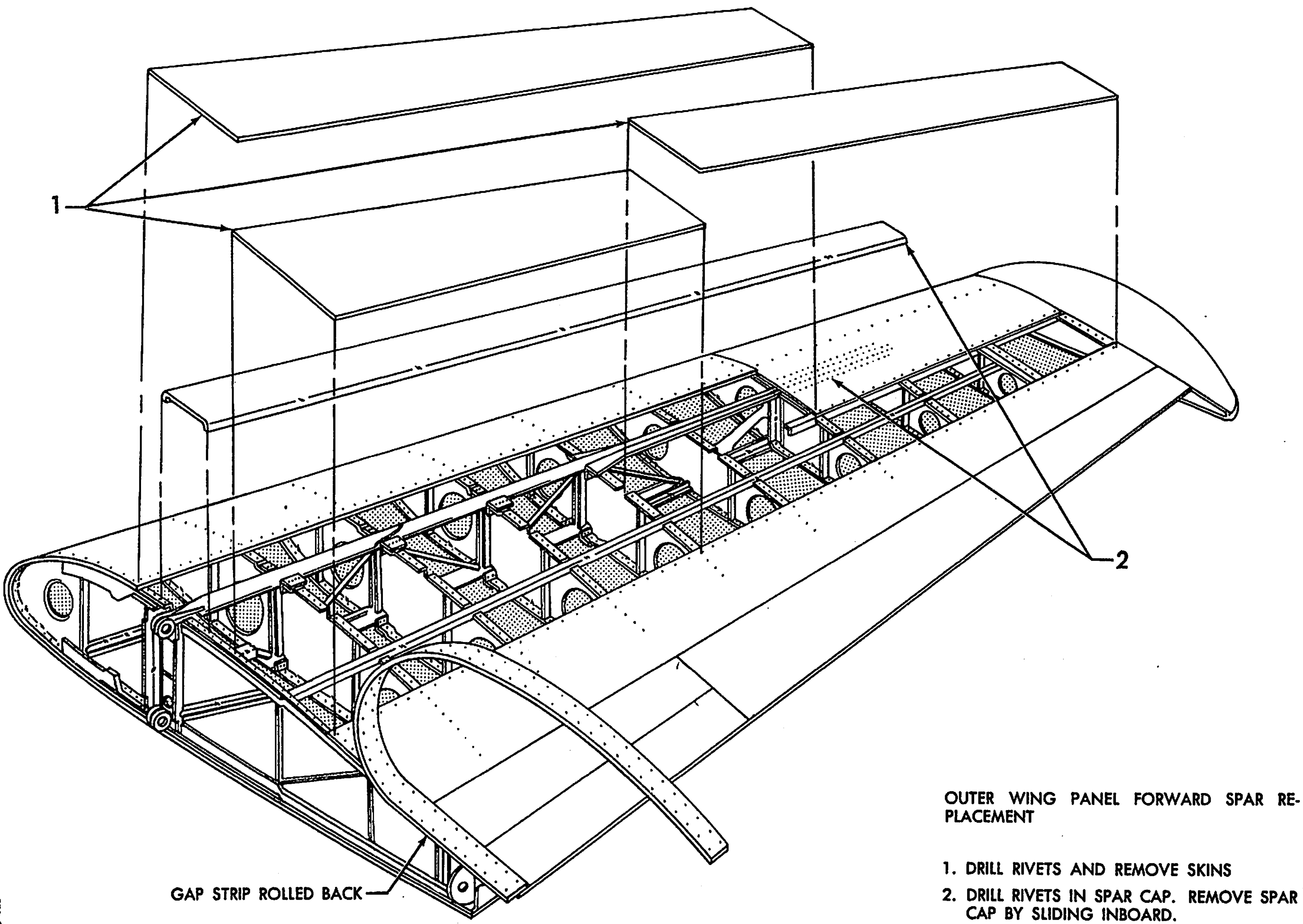
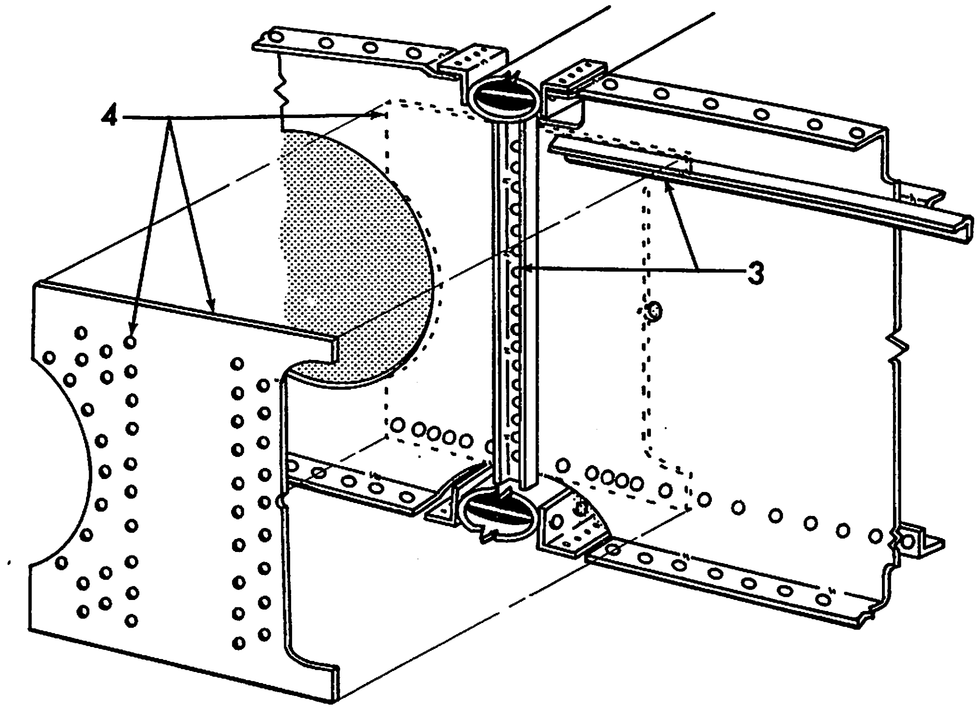


Figure 2-3A. Outer Wing Panel Front Spar Replacement (Sheet 1 of 6 Sheets)

CAUTION: DRILL SPLICE PLATE RIVET PATTERN BEFORE RIBS ARE CUT.

3. DRILL RIVETS IN U-CHANNEL AND REMOVE ANGLE SO SPLICE PLATES CAN BE LOCATED ON RIBS. (INDICATIVE -2 SPLICE PLATE ON NUMBER 2 RIB; -4 SPLICE PLATE ON NUMBER 3 RIB; -6 SPLICE PLATE ON NUMBER 4 RIB AND -8 SPLICE PLATE ON NUMBER 5 RIB) (SEE PAGE 41.)
4. LOCATE SPLICE PLATES ON RIBS AND DRILL IN RIVET PATTERNS ON RIBS. TRIM SPLICE PLATES TO CLEAR DE-ICER TUBING.



5. FILLERS (MADE OF .040 SHEET OR STRIP CLAD 24ST-3 AL F.E.D. QQ-A-362 COND. T3) TO BE INSTALLED AT INSTALLATION OF SPLICE PLATES. (SEE PAGE 37, STEP 5)
6. REINSTALL ANGLE STIFFENERS AND RIVET TO SPLICE PLATES AFTER SPLICE PLATES ARE INSTALLED. (SEE PAGE 40, STEP 16)

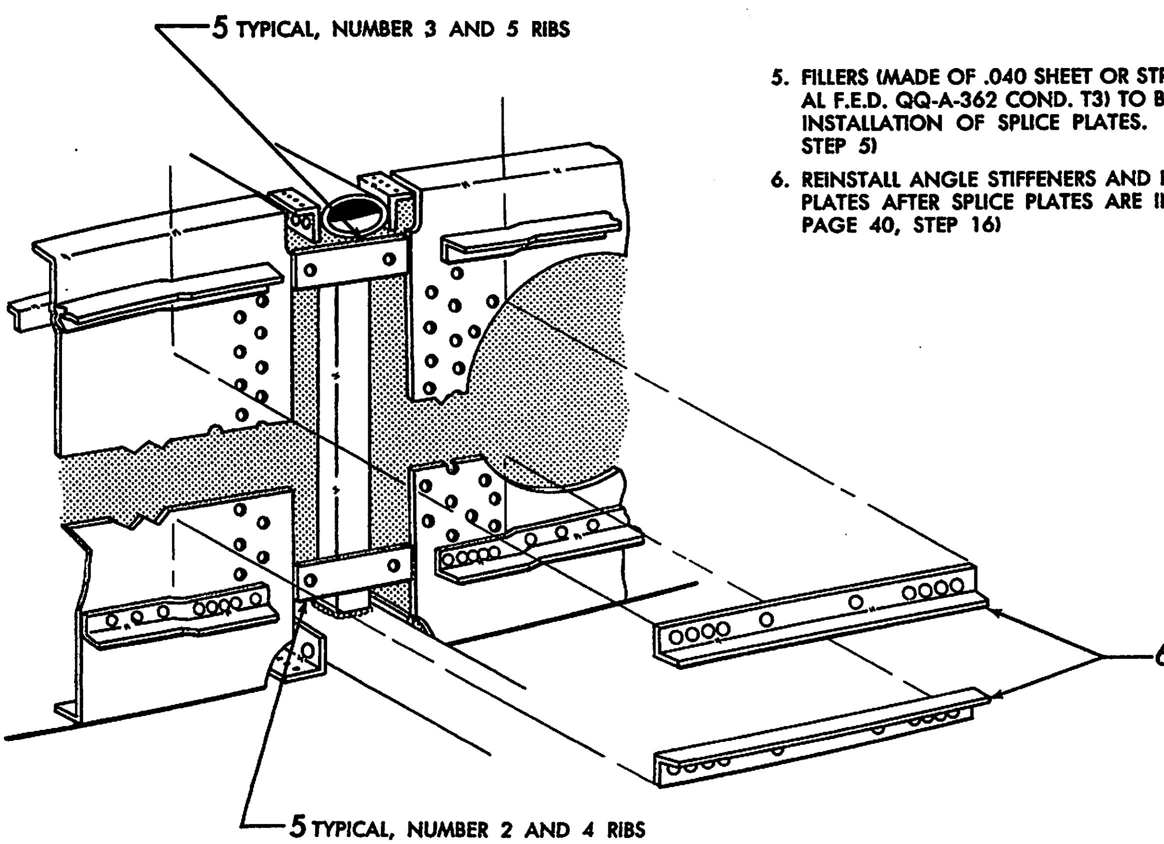
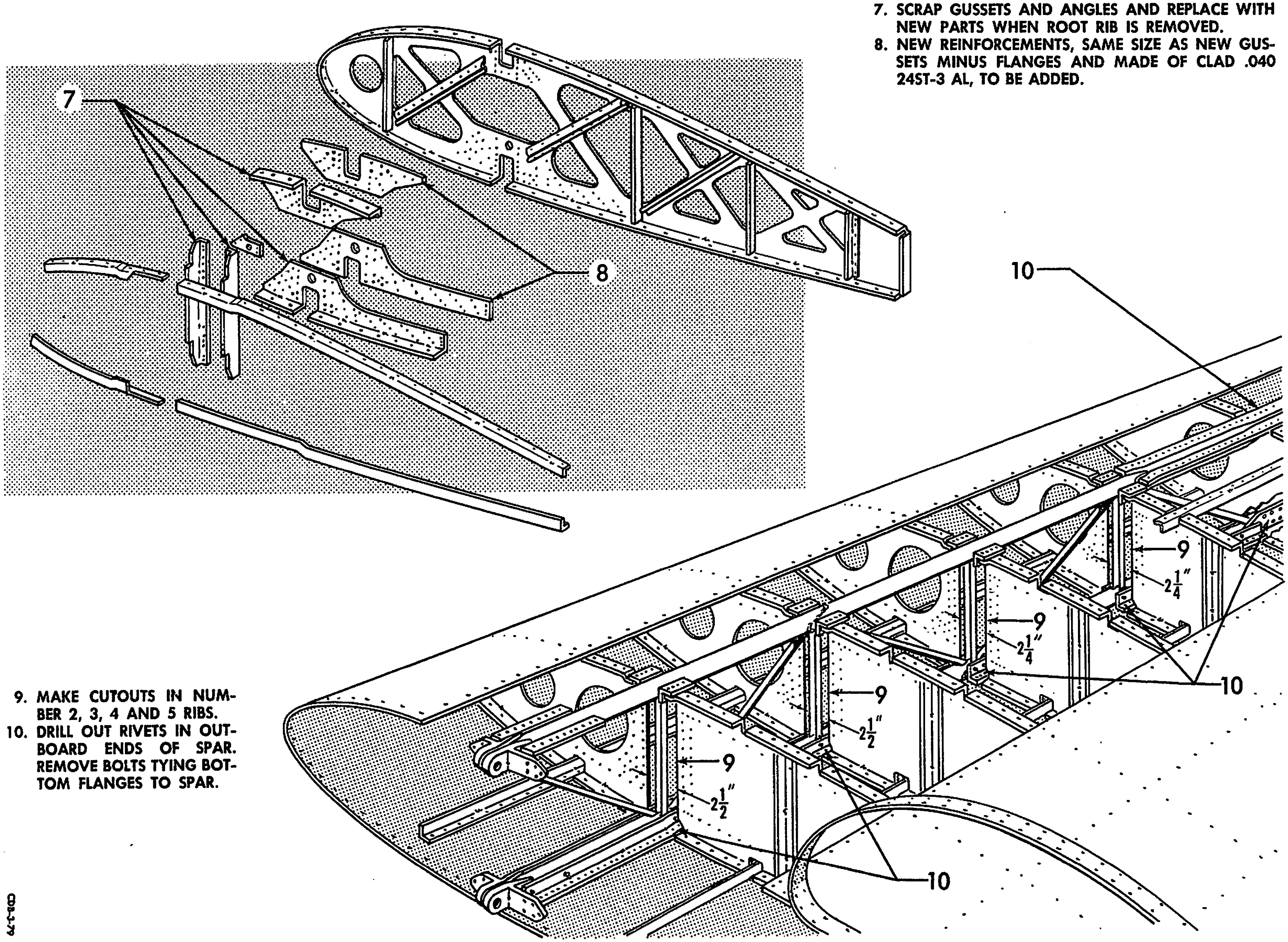


Figure 2-3A. Outer Wing Panel Front Spar Replacement (Sheet 2 of 6 Sheets)

7. SCRAP GUSSETS AND ANGLES AND REPLACE WITH NEW PARTS WHEN ROOT RIB IS REMOVED.
8. NEW REINFORCEMENTS, SAME SIZE AS NEW GUSSETS MINUS FLANGES AND MADE OF CLAD .040 24ST-3 AL, TO BE ADDED.



9. MAKE CUTOUTS IN NUMBER 2, 3, 4 AND 5 RIBS.
10. DRILL OUT RIVETS IN OUTBOARD ENDS OF SPAR. REMOVE BOLTS TYING BOTTOM FLANGES TO SPAR.

Figure 2-3A. Outer Wing Panel Front Spar Replacement (Sheet 3 of 6 Sheets)

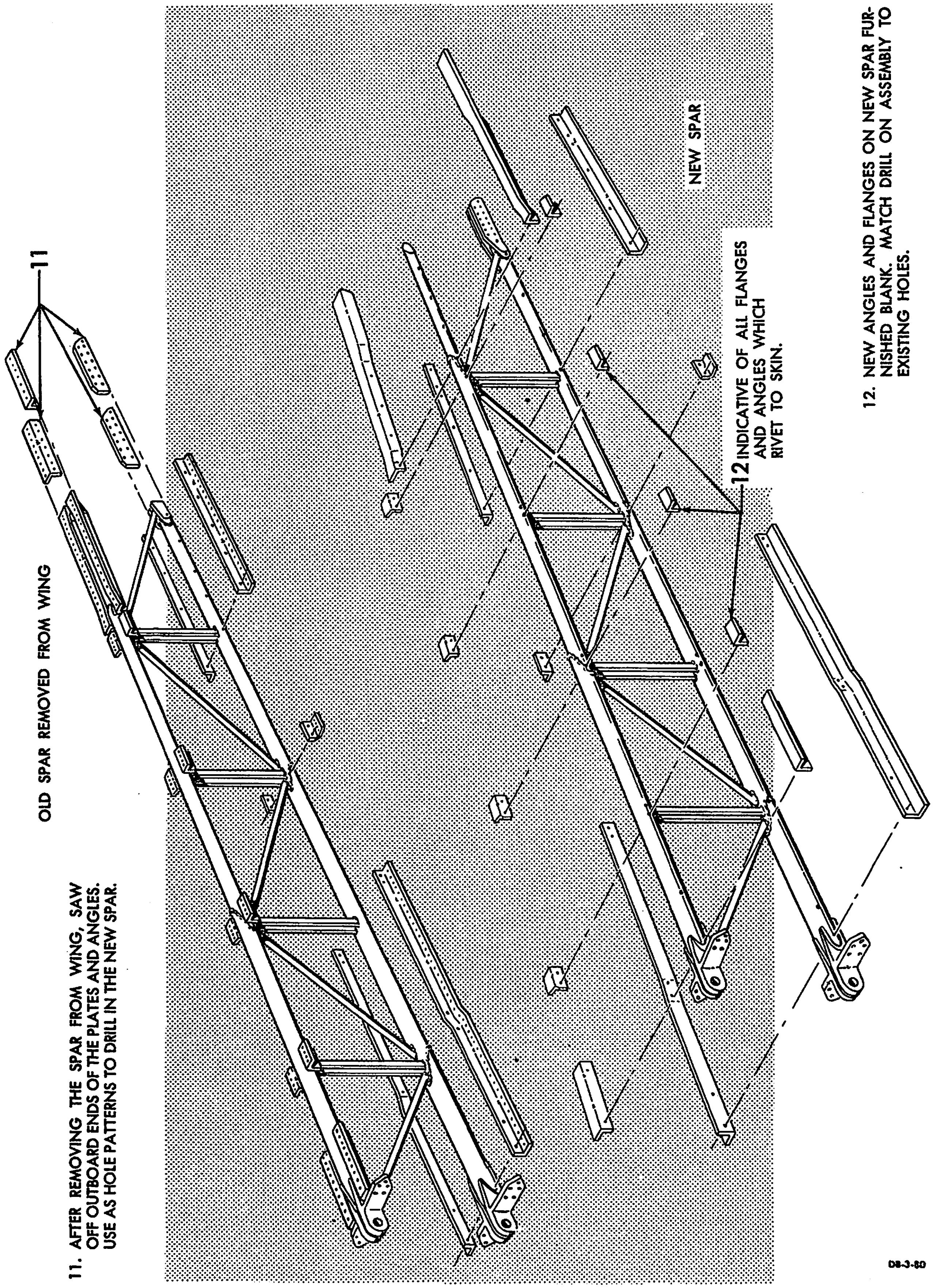


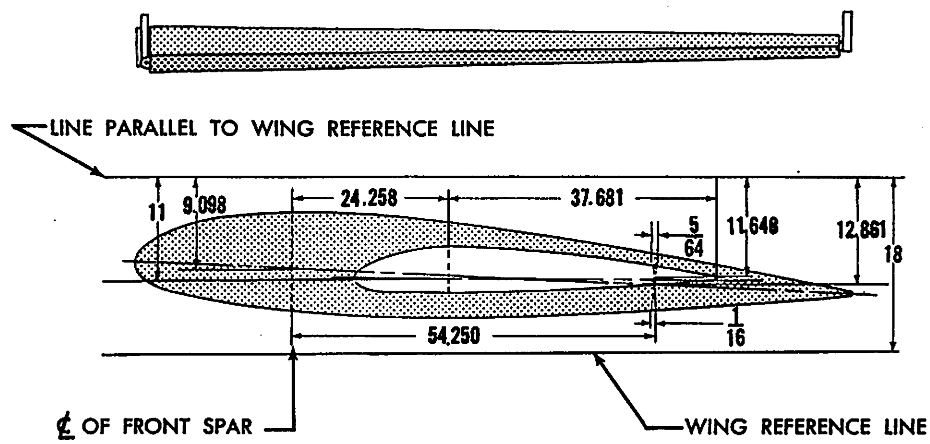
Figure 2-3A. Outer Wing Panel Front Spar Replacement (Sheet 4 of 6 Sheets)

13. INSTALL NEW SPAR IN WING. INSTALL ROOT RIB. BUTTON DOWN ALL RIVETS IN ROOT RIB AND SKIN.
CAUTION: LEVELING JIG MUST BE INSTALLED BEFORE PROCEEDING WITH FOLLOWING STEPS.

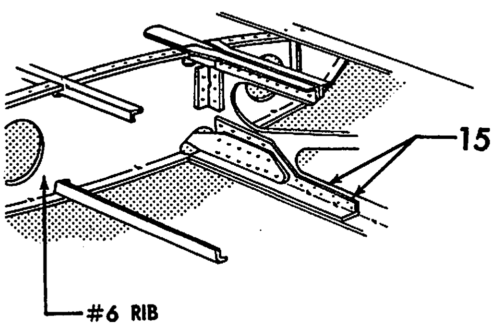
14. PUT WING IN JIG FOR MATING INBOARD AND OUTBOARD SPARS. USE HOLEFINDER FOR LOCATING TWO HOLES IN PLATES AND ANGLES AT OUTBOARD END OF NEW SPAR. REMAINDER OF HOLES TO BE DRILLED FROM OLD PLATES AND ANGLES.

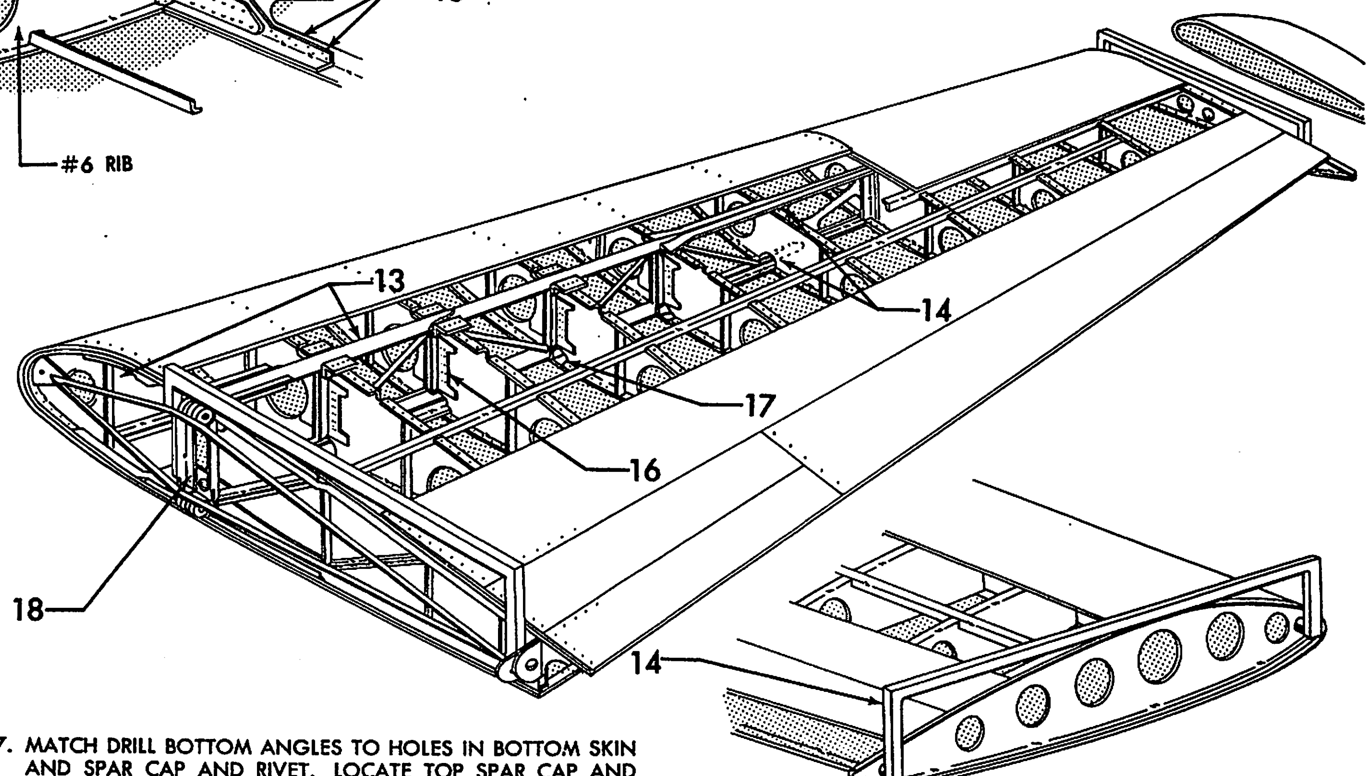
15. ANGLES (FWD. AND AFT SIDES) ARE USED ONLY WHEN HOLES IN LOWER PLATES MISALIGN WITH EXISTING HOLES IN OUTER FRONT SPAR. AD6 RIVETS USED IN PLACE OF AD5 RIVETS IN CASE OF SLIGHT MISALIGNMENT.

SUGGESTED JIG TO BE CONSTRUCTED TO INCORPORATE 3% TWIST



16. INSTALL SPLICE PLATES ON RIBS
-2 SPLICE PLATE ON #2 RIB
-4 SPLICE PLATE ON #3 RIB
-6 SPLICE PLATE ON #4 RIB
-8 SPLICE PLATE ON #5 RIB
ALIGN RIBS WITH SPAR AND MATCH DRILL SPLICE PLATE TO U-CHANNEL ON SPAR AND RIVET.





17. MATCH DRILL BOTTOM ANGLES TO HOLES IN BOTTOM SKIN AND SPAR CAP AND RIVET. LOCATE TOP SPAR CAP AND ALL TOP SKINS. AN470 RIVETS MAY BE REPLACED WITH NEXT SIZE LARGER IF NECESSARY. DO NOT REDIMPLE DIMPLED SKINS TO NEXT LARGER SIZE. BACK DRILL TOP ANGLES ON SPAR TO MATCH EXISTING HOLES IN SKIN AND SPAR CAP.

18. BACK DRILL ROOT RIB TO SPAR AND INSTALL BOLTS.

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Figure 2-3A. Outer Wing Panel Front Spar Replacement (Sheet 5 of 6 Sheets)

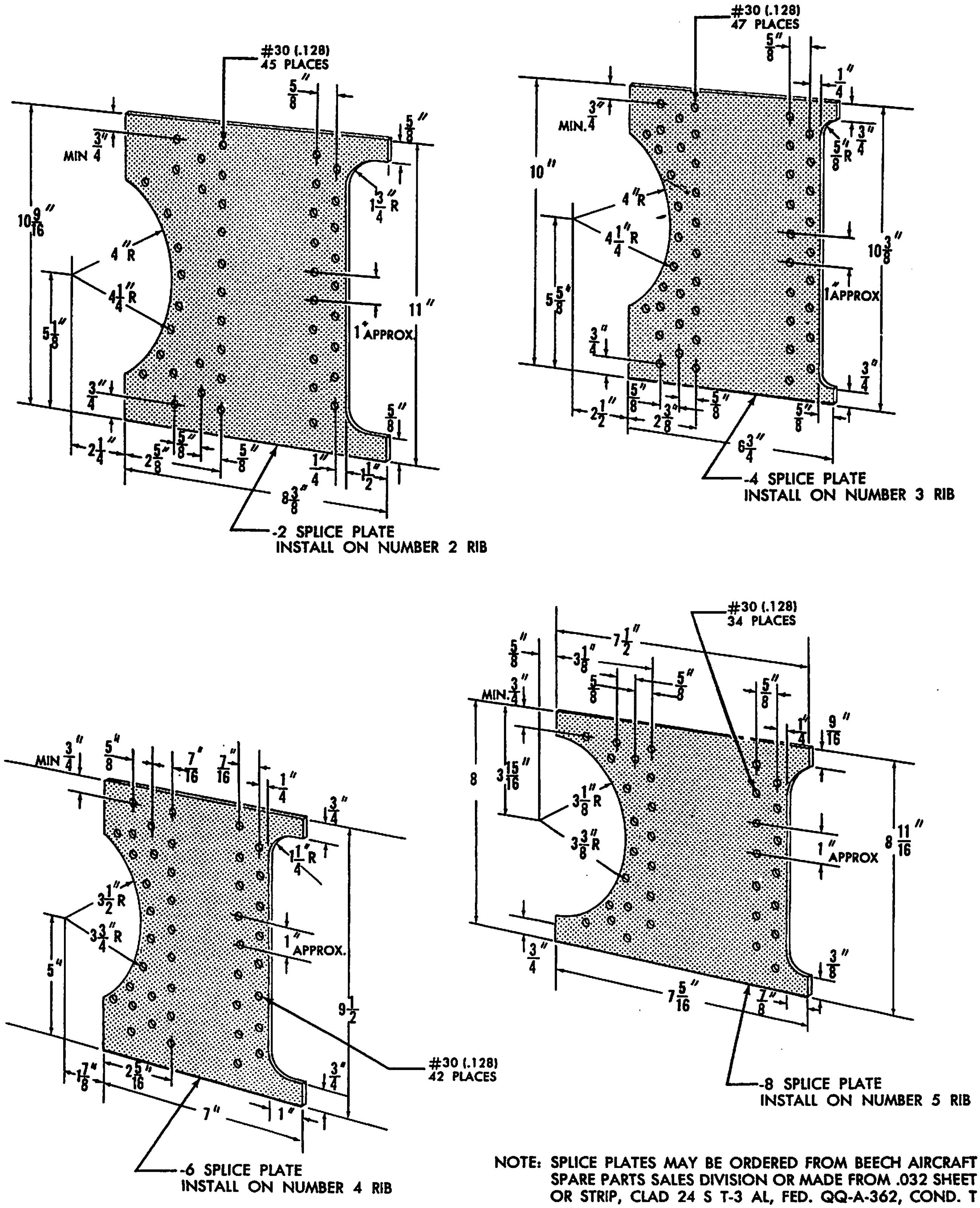

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Figure 2-3A. Outer Wing Panel Front Spar Replacement (Sheet 6 of 6 Sheets)

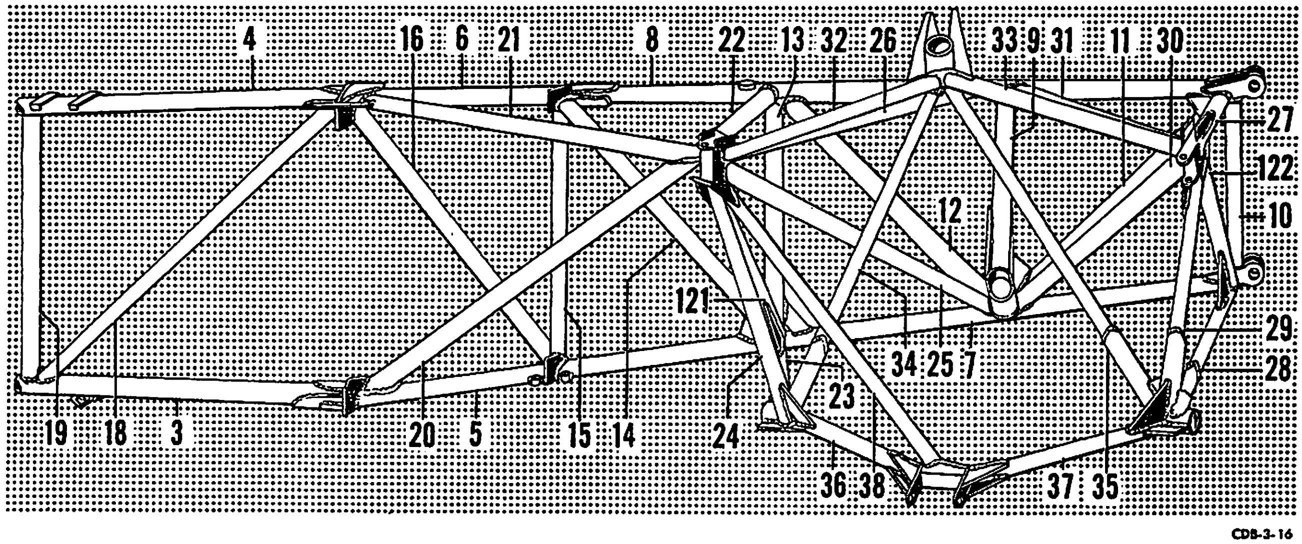

Figure 2-4. Truss Tube Members, Numbered

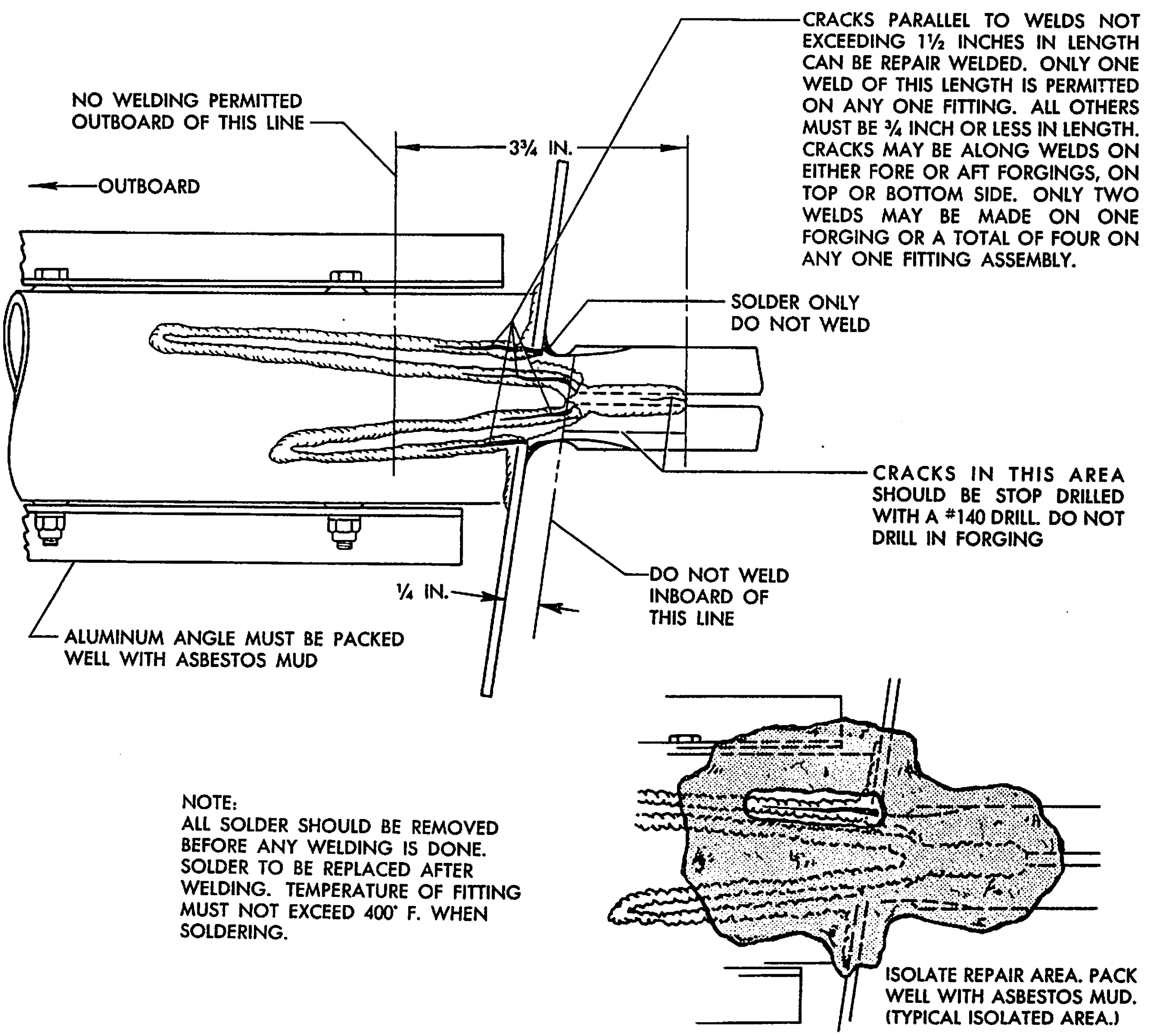

Figure 2-4A. Outer Wing Panel Spar Fitting

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neutral flame directed on the crack alone or on the bottom of the ground-out area.

NOTE

The No. 22 tip is a Smith designation tip, having an orifice size of .0310 (the size of a 68 drill).

d. Continue preheating until the point directly below the cone of the flame attains black heat or purple oxide rings extend 1/4 inch from the crack. This will indicate sufficient preheating of the area.
e. Arc weld immediately, using the method described in paragraph 2-63A. After the red heat disappears, peen the bead with a ball peen hammer which has a 1/8-inch head radius.
f. Before the weld has cooled below 350°F (light straw color), post-heat by concentrating the flame from a No. 22 tip (Smith orifice) along the center of the new bead.
g. Continue post heating until the bead reaches black heat (950°F/1000°F) or purple oxide rings extend 1/4 inch from the bead (500°F).
h. Allow the repair to cool in still air.
i. Magnaflux after the repair has cooled.

2-64. REPLACEMENT AND REPAIR OF WELDED CLUSTERS.

2-65. Repairs to the several welded clusters may be made only within the limits prescribed in the following paragraphs. Those clusters which are replaceable may be replaced with clusters from a salvage truss, as directed in the individual paragraphs and illustrations. In all cases, however, care must be taken to remove the damaged cluster and the cluster from the salvage truss in exactly the same manner. See figure 2-4 for tube identification.

2-66. LOWER SLIDE TUBE CLUSTER.

2-67. This cluster is not replaceable and is repairable only to the extent shown in figure 2-5, and within the following limits: Cracks on bottom of -7 may be repaired within the same limits noted on top side of tubes (figure 2-5) except that only one repair may be made in any given cross section; i.e. repairs must be staggered at least one inch.

2-68. WING SPAR BOLT FITTING CLUSTERS.

2-69. These clusters are not replaceable and are repairable only within the limits shown in figure 2-6. The ends of the wing outboard panel spar caps, where the wing hinge bolt forgings are welded in, are filled with solder. Linoil seepage at these points will require very careful inspection to determine whether the weld or tube is cracked or the solder merely has parted from the joint. All solder should be removed before welding and replaced when weld is finished. Temperature of the fitting during soldering must not exceed 204°C (400°F).

2-69A. Weld repairs may be made to cracks appearing in the outboard wing panel front spar, where the spar bolt fitting is welded to the spar cap, if they are no longer than 1-1/2 inches and run along the weld; however, only one weld of this length is permitted on any one fitting. All other welds must be 3/4 inch or less. Cracks along welds on either fore or aft forgings on top or bottom sides, may be repaired, but only two welds may be made on one forging or a total of four on any one fitting assembly. Repairs to the fittings and their attachments must be made in accordance with the instructions shown in figure 2-4A.

2-70. ENGINE MOUNT FITTINGS.

2-71. Cracks in the engine mount fittings may be repaired only to the extent shown in figure 2-7, and within the following limits:

WARNING

When repairing the heat treated tubes by welding, a partial destruction of the strength occurs. Therefore only those clusters or tubes on which repairs are approved will be discussed. It may be considered that any tube or cluster not covered is not repairable. Do not exceed limits set forth for each repair.

a. Cracks in the bends of the fitting center piece may be welded entirely across the center piece. See 1, figure 2-7.
b. Cracks between the fitting and tubes may be welded when crack is transverse. Tube may be welded to an extent of 5/8 inch. See 2, figure 2-7.
c. Cracks between the fitting lug pieces and the fitting center pieces may be welded to the extent of 5/8 inch. See 3, figure 2-7.
d. Cracks between fittings or gussets and tubes may be welded when the crack is parallel to the center line of the tube and is not more than 1 inch long. See 4, figure 2-7.

NOTE

A second weld in the same fitting should not be made until the first weld has cooled, to minimize heat transfer.

e. Upon the discovery of a crack, either by visual inspection or magnaflux, grind to determine extent of crack. Use a rotary file with a 1/8-inch radius, being careful not to exceed limits prescribed for repair. If crack continues beyond repair limit, replace-

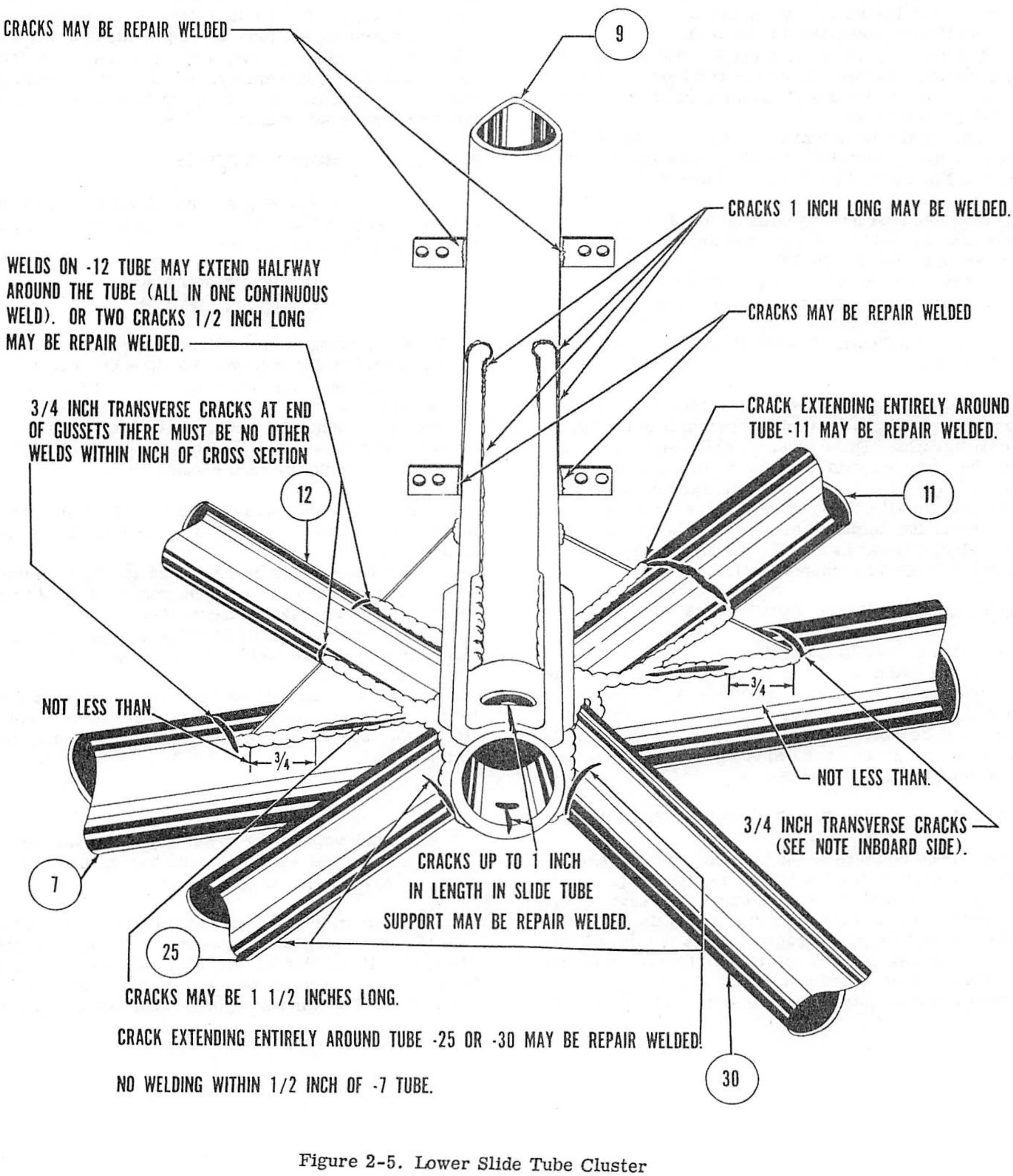
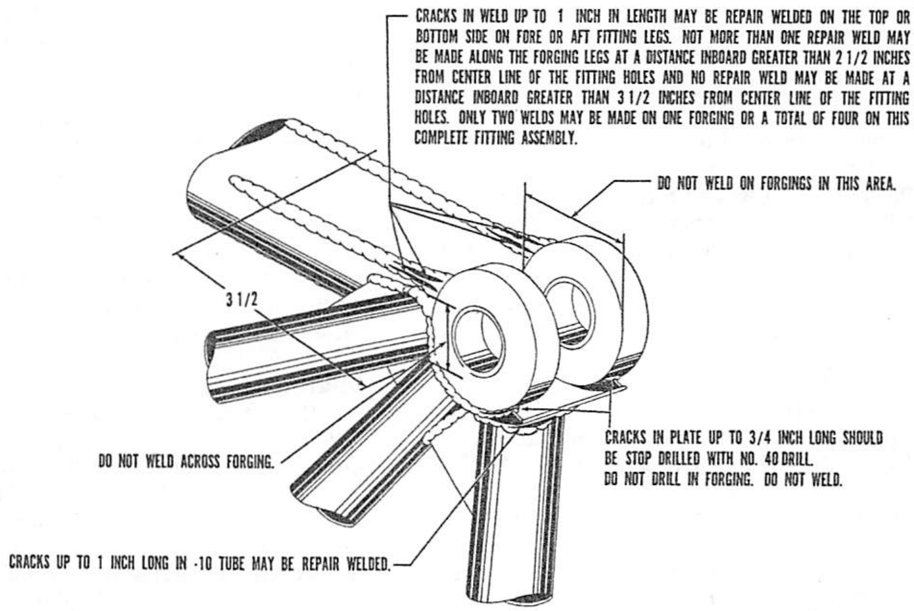
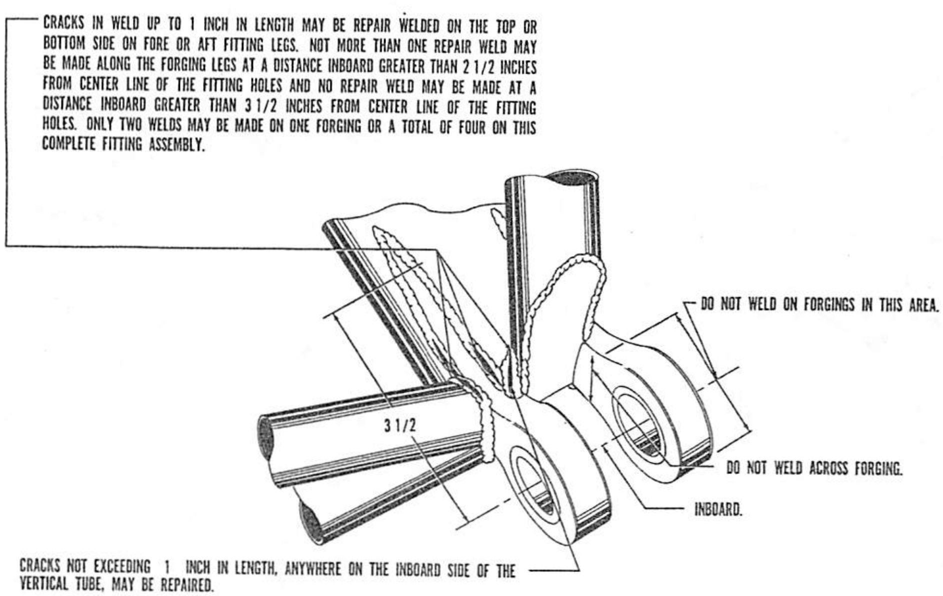


Figure 2-5. Lower Slide Tube Cluster



NOTE: A SECOND WELD IN THE SAME FITTING SHOULD NOT BE MADE UNTIL FIRST WELD HAS COOLED



NOTE: CARE MUST BE EXERCISED IN GRINDING OUT CRACKS TO AVOID EXCEEDING CRACK EXTENT AND GRINDING INTO VOID. A SECOND WELD IN THE SAME FITTING SHOULD NOT BE MADE UNTIL FIRST WELD HAS COOLED

Figure 2-6. Wing Spar Bolt Fitting Clusters ment of tube or cluster may be necessary. Careful distinction must be made between a crack and an actual void at the end or bottom of weld, where the weld has not penetrated. If possible, repair weld should set overhight before magnafluxing.

2-72. UPPER SLIDE TUBE CLUSTER.

2-73. Replacement of damaged upper slide tube cluster requires the cutting and splicing of tubes -32, -33, -34, and -35. See figure 2-8. Use the following procedure:

a. The slide tube must be removed. Remove the bolts in the fittings at each end of the slide tube and slide the tube through the top fitting.

NOTE

In most cases where the upper slide cluster is damaged, the slide tube will be damaged beyond repair. In such cases, removal of the tube will be simplified by sawing it in half.

b. Locate the position for cutting the tubes by referring to figure 2-8.

c. Place the splice tubes on the tubes in the truss and slide down past cut off ends.

d. Install the replacement fitting and slide the splice tubes equidistantly over the joints. With the fitting thus held in position by the sleeves, install a slide tube and bolt into place.

e. After tacking the fitting in place, weld the ends of the splice sleeves, making but one weld at a time on each sleeve. Magnaflux after welding.

f. Install the landing gear. Check travel for complete retraction of gear.

g. If necessary dress off the upper slide tube fitting to allow full upward travel of the slide, or tack a spacer on the fitting to limit upward travel.

2-74. LOWER ENGINE MOUNT FITTING CLUSTER.

2-75. The lower engine mount fitting cluster may be replaced. To do so proceed as follows:

a. Obtain a replacement fitting from a salvage truss being careful to remove it and the damaged cluster in an identical manner.

b. Removal of the cluster will require cutting tubes -36, -37 and -38. See figure 2-9 for location of the cuts on -38 and figure 2-10 for -36 and -37.

c. Splicing of the -38 tube depends on the extent of the damage. If the tube has been buckled due to damage of the lower engine mount fitting, the tube should be cut at AA and BB and spliced with the long splice as shown in figure 2-9. If tube is not damaged, cut at BB and splice with a short splice tube.

d. With the damaged fitting removed and a replacement

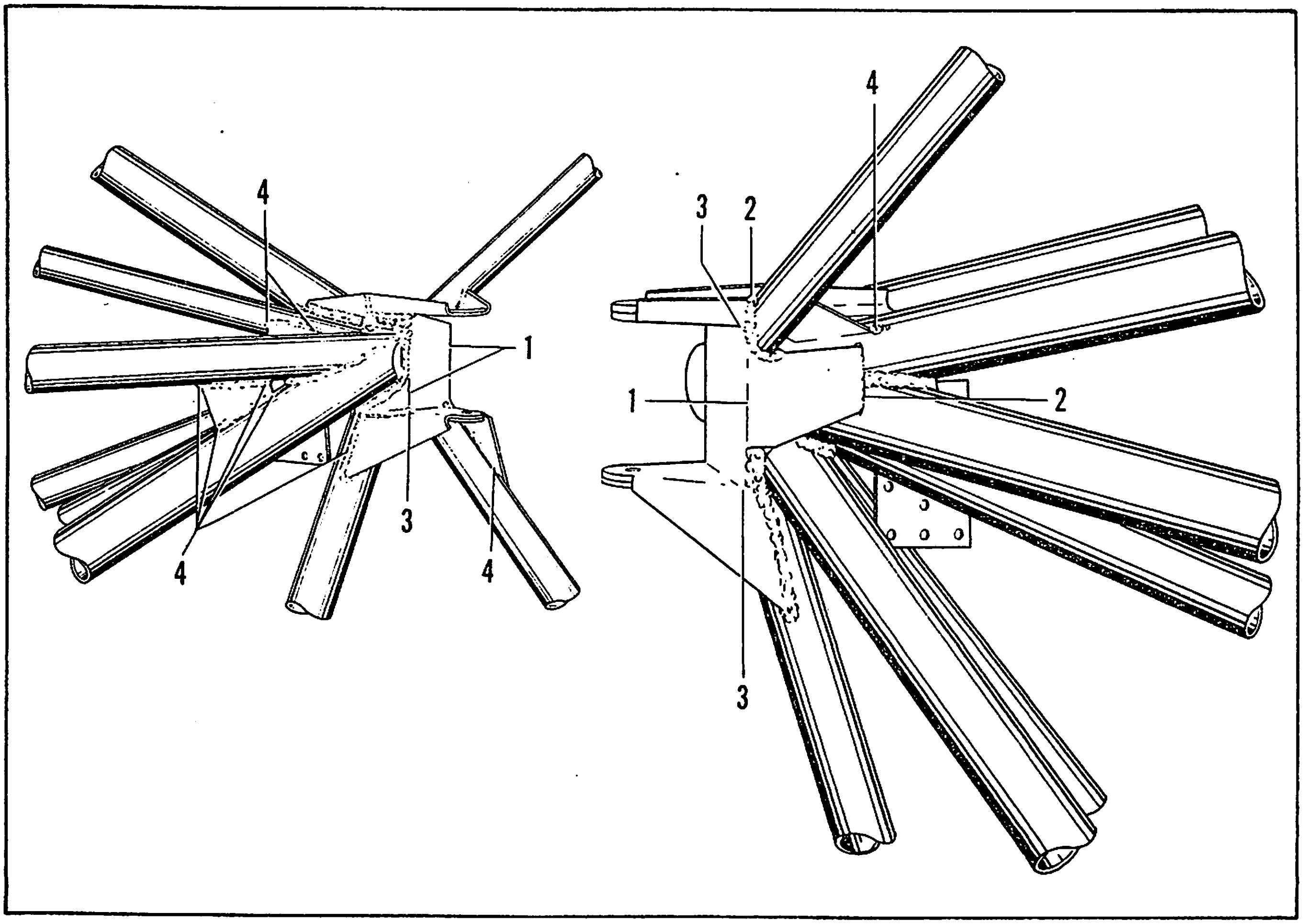


Figure 2-7. Engine Mount Fitting Repairs

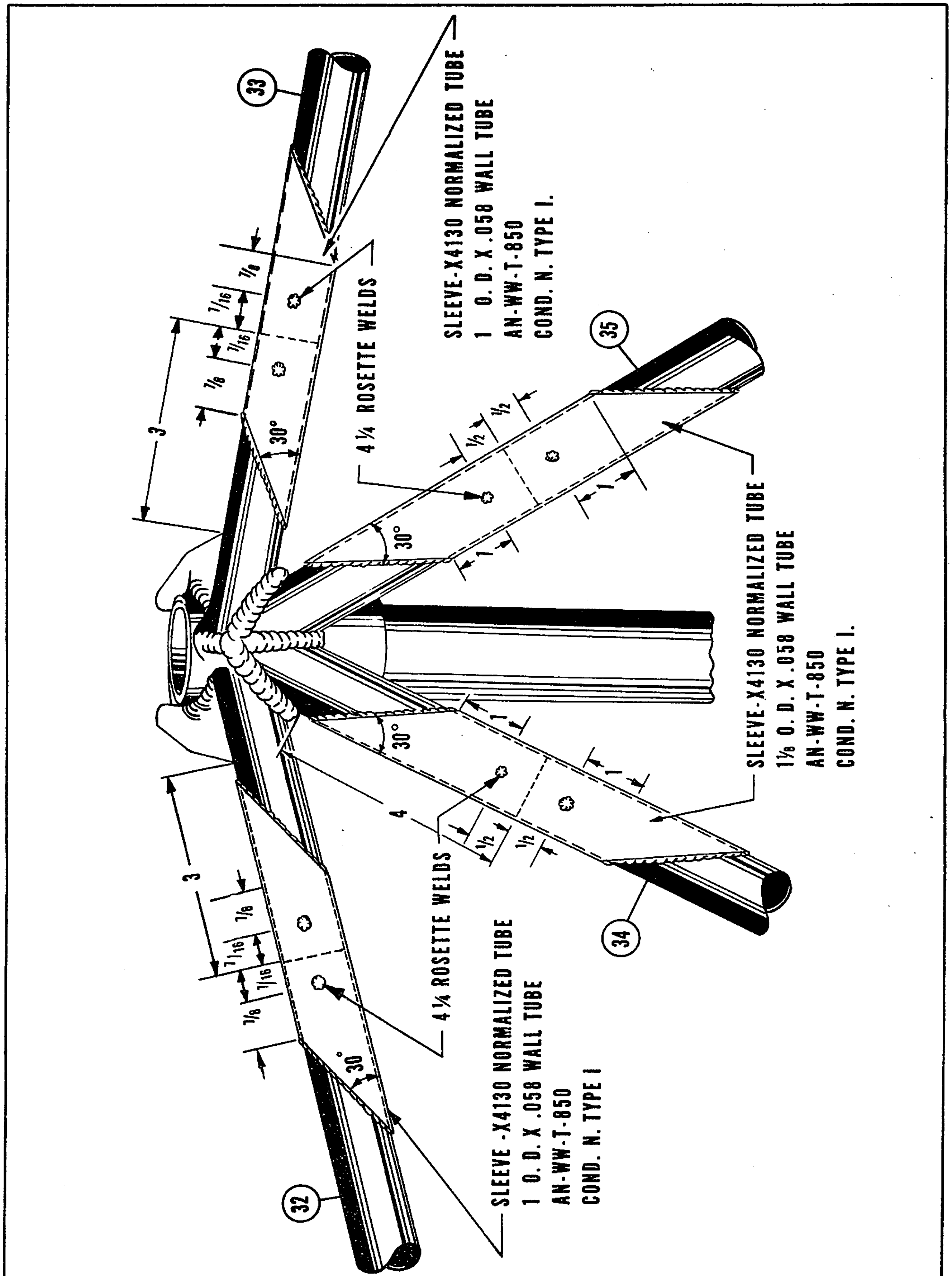


Figure 2-8. Upper Slide Tube Cluster

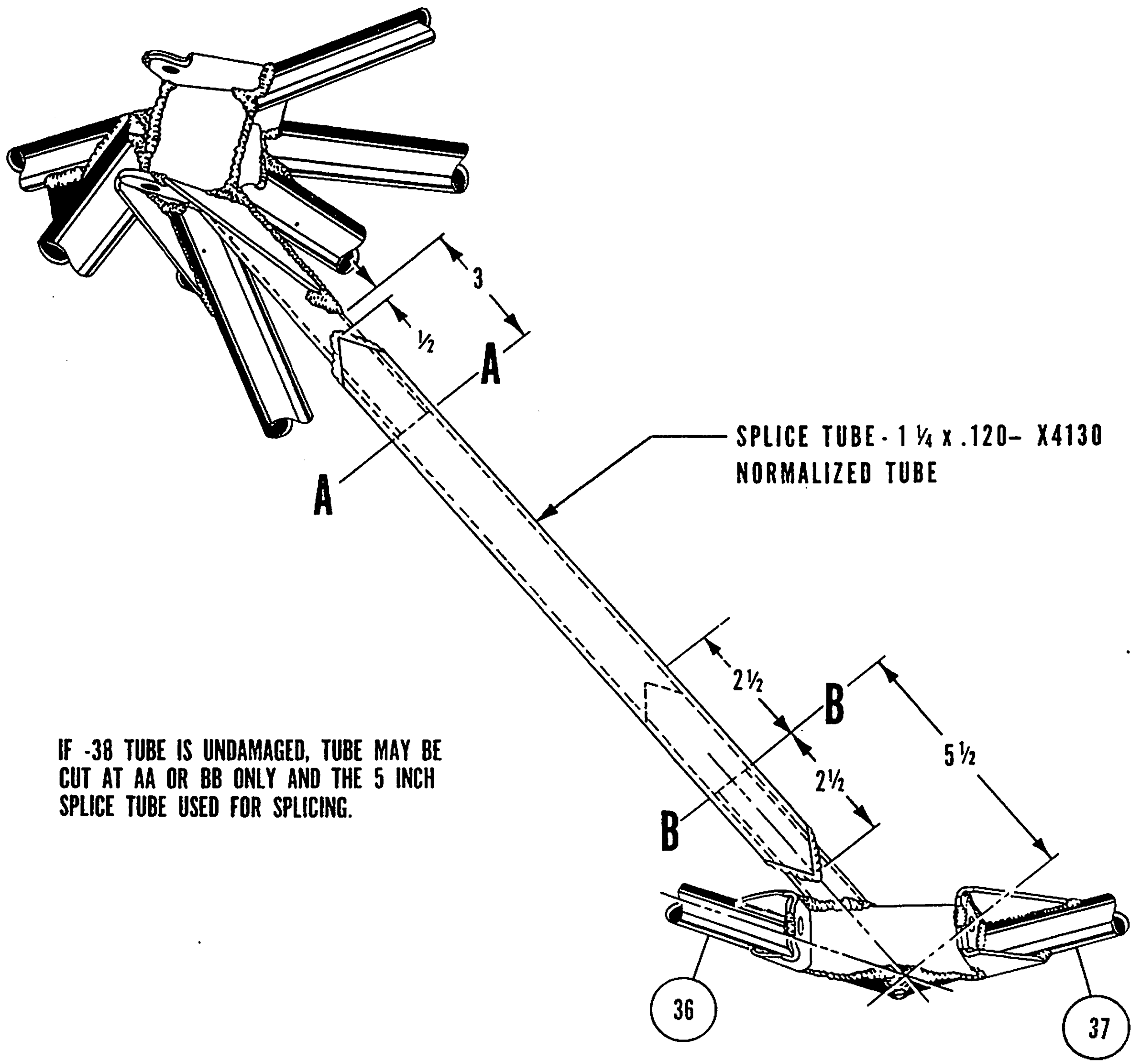


Figure 2-9. Splicing-38 Tube

fitting ready to install, slide the splice tubes for -36, -37, and -38 on their respective stubs on the fitting.

e. Move the fitting into its correct position and slide splice tubes equidistantly over the tube cuts.

f. Install an engine mount to assist in locating and steadying the position of lower fitting.

g. Obtain the fore and aft locations. This is done by measuring from an axis through the center of the 7/16-inch bolt hole and the landing gear hinge fitting. This measurement is centered as shown in figure 2-11. The true dimension is 13-9/64 inches, plus 1/32-inch. This can be obtained by inserting a one-inch round bar through the landing gear hinge fitting, measuring between the bar and the 7/16-inch bolt and adding the sum of the radii of the bar and bolt.

h. When the fitting is properly located, tack weld to hold it in position.

i. Remove engine mount and one-inch bar. Weld ends of splice tubes, making but one weld at a time on each sleeve. Magnaflux after welding.

NOTE

If the lower Lord rubber bushing in the engine mount is worn to the extent that bolt will not center in the bushing, replace it before using mount as a jig.

2-76. INBOARD LANDING GEAR HINGE FITTING CLUSTER.

2-77. To remove a damaged inboard landing gear hinge cluster, proceed as follows:

a. Remove the cluster from the aircraft and from the salavage truss. See paragraph 2-65.

b. Removal of the inboard landing gear hinge fitting requires cutting the -23, -24, -34, and -36 tubes. The location for cutting -23 at AA is shown in figure 2-12, for -24 at BB figure 2-13, for -34 at BB figure 2-14, and -36 figure 2-10.

c. Obtain material for splice tubes and cut them as shown in their respective figures.

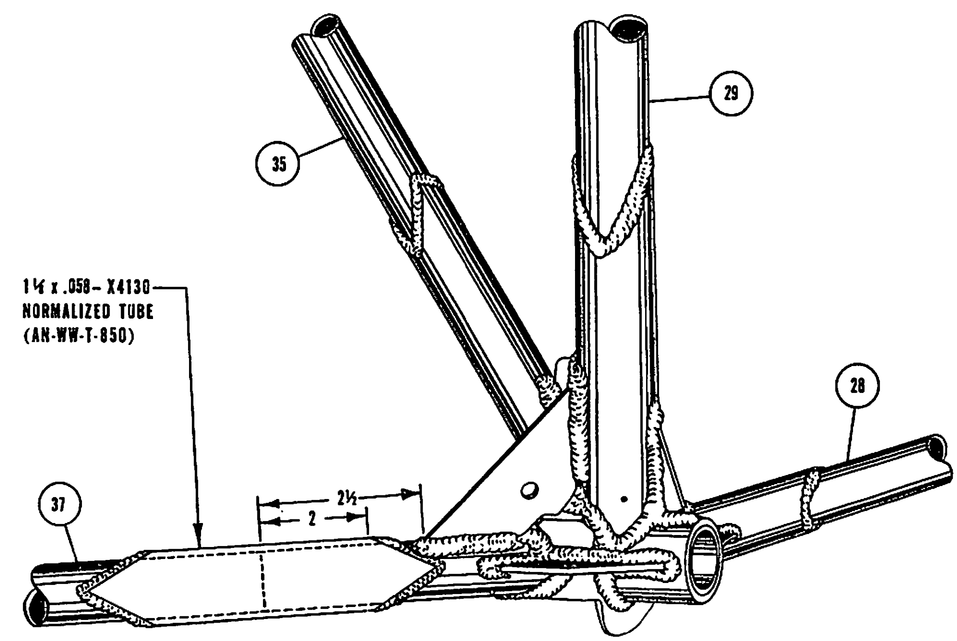
d. Mark splice tubes for proper location of splice gap. Also mark tubes to properly locate sleeves.

e. Cut the splice tubes for the -23, -24, -34, and -36 splice tubes as shown in their respective illustrations.

f. Place the -23, -24, and -34 splice tubes on the replacement cluster. The -36 splice tube must be placed on the truss portion of the tube.

g. Position the replacement cluster and slide the splice sleeves into place, aligning with marks in step d. With the cluster now loosely installed, install an engine mount to keep the lower engine mount fitting in place.

h. Install the landing gear. If the landing gear bush-



NOTE: THIS SPLICE IS GOOD FOR BOTH -36 AND -37 TUBES AND CAN BE MADE ON EITHER END OF THE TUBE. IN SPLICING AT THE END NEXT TO THE ENGINE MOUNT FITTING, THE 2½" DIMENSION WILL BE FROM THE FARTHEST POINT UP WHERE THE WELD ATTACHES THE GUSSET TO THE TUBE.

Figure 2-10. Splicing -36 and -37 Tubes

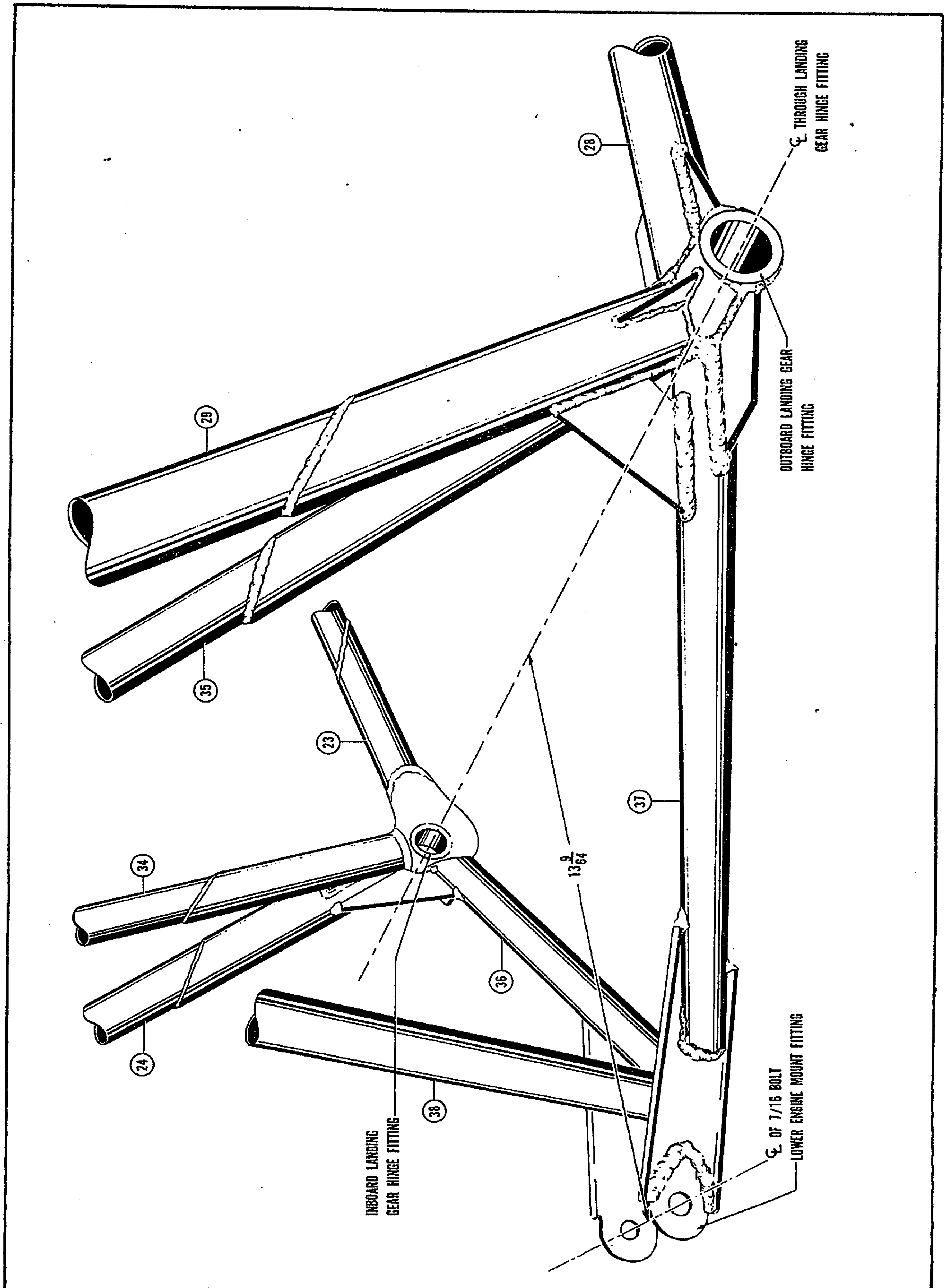


Figure 2-11. Locating Lower Engine Mount Cluster

ings are worn, install new ones.

i. Adjust wheels for "O" caster and camber. When hinge joint cluster is properly located, tack weld the joints in the following order: -34, -24, -23, and -36.

j. Remove landing gear and engine mount. Weld splices in the following order: -23, -24, -36, and -34.

2-78. OUTBOARD LANDING GEAR HINGE JOINT CLUSTER.

2-79. To replace a damaged outboard landing gear hinge joint cluster proceed as follows:

a. Obtain a salvage cluster and remove damaged cluster. See paragraph 2-65. This will require cutting tubes -28, -29, -35, -37. Cut tubes -28 at BB figure 2-15, -29 at BB figure 2-13, -35 at BB figure 2-14, and -37 as in figure 2-10.

b. Cut and mark splice sleeves and mark tubes for sleeve location.

c. If landing gear bushings are worn, replace them. Install landing gear to provide means for aligning cluster. Adjust wheels for "O" caster and camber, moving free hinge point to obtain proper alignment.

CAUTION

While aligning the landing gear, be sure that aircraft is completely supported on jacks and that both gears are fully extended.

d. Tack weld the hinge point after it is properly aligned with respect to caster and camber. Tubes should be tacked in the following order: -35, -29, -28, and -37.

e. Remove landing gear.

CAUTION

Remove vent plugs on tubes being welded. Failure to do so may result in an explosion while welding.

f. Make final weld on tubes in the following order: -28, -29, -37, and -35. Make but one weld at a time on each tube. Welding both ends of a splice at one time will cause axial expansion on the tube. Before making second weld on tube, allow it to cool. After completion of weld, magnaflux.

g. Replace vent plugs.

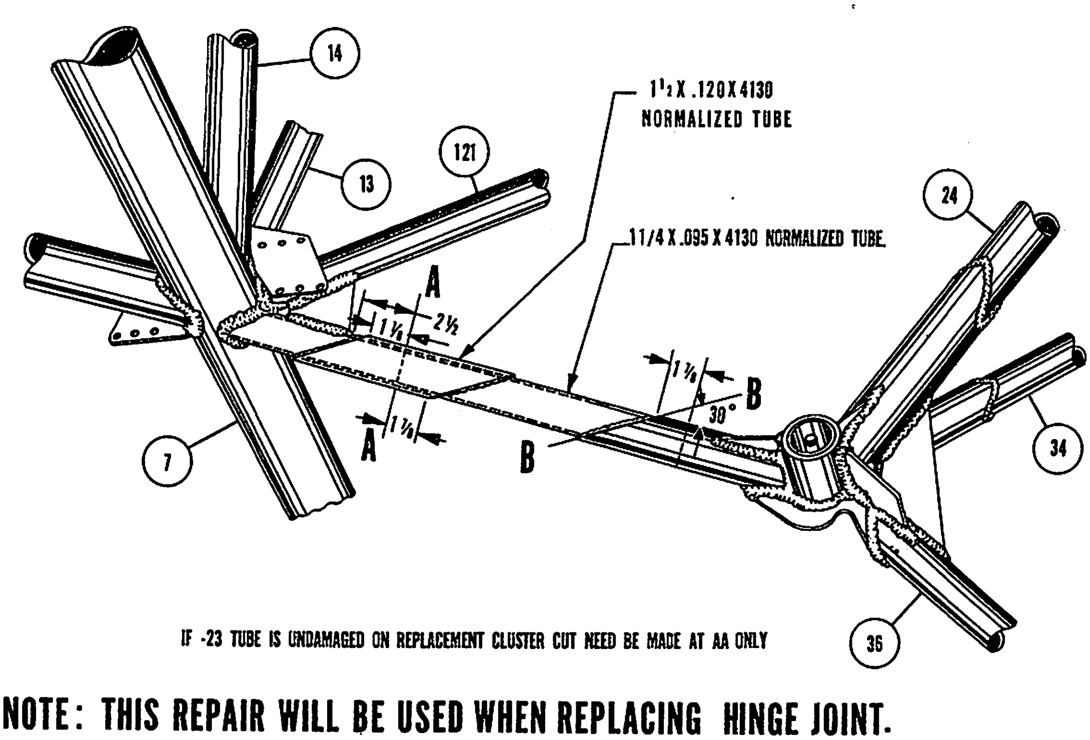


Figure 2-12. Splicing -23 and -28 Tubes

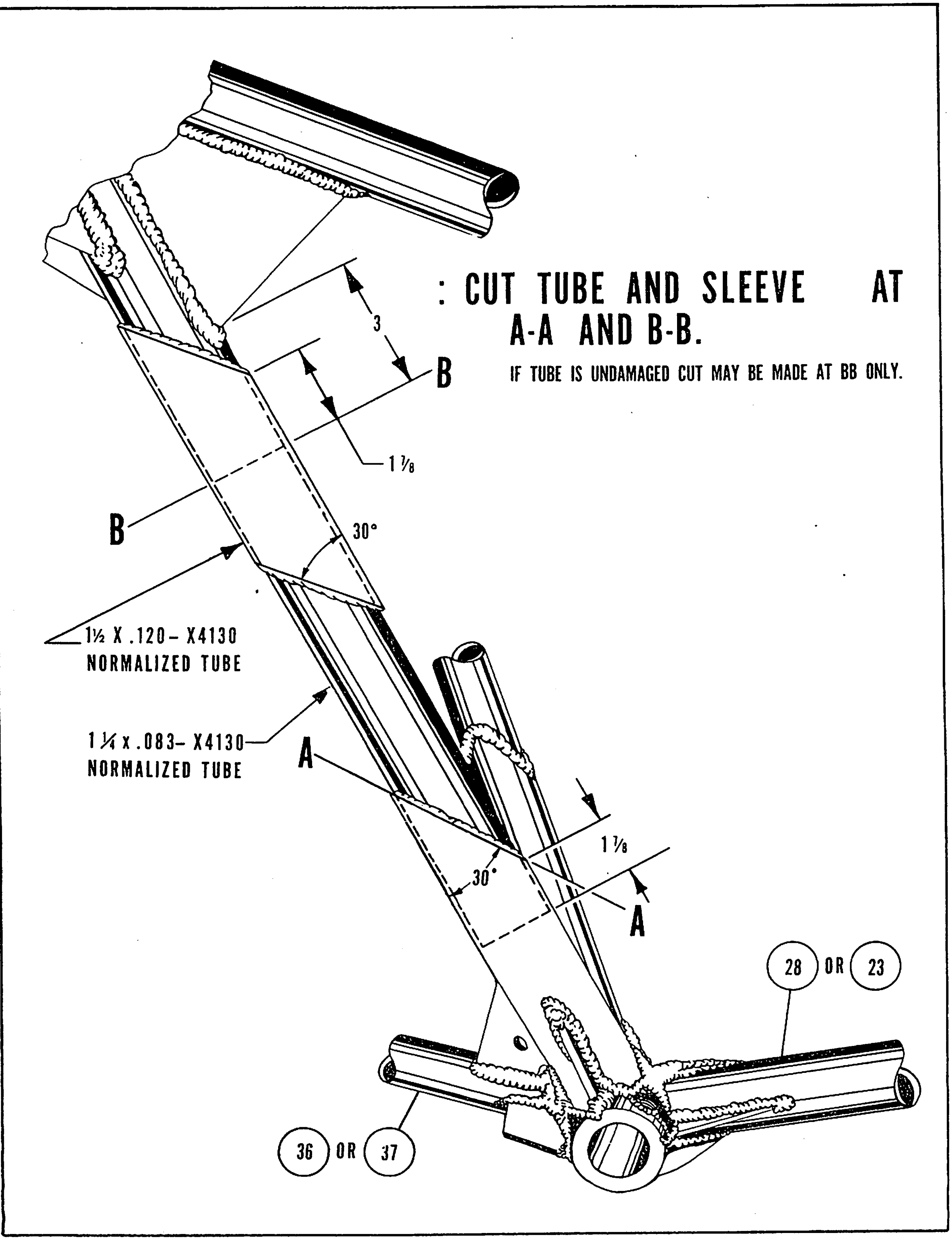


Figure 2-13. Splicing -24 and -29 Tubes

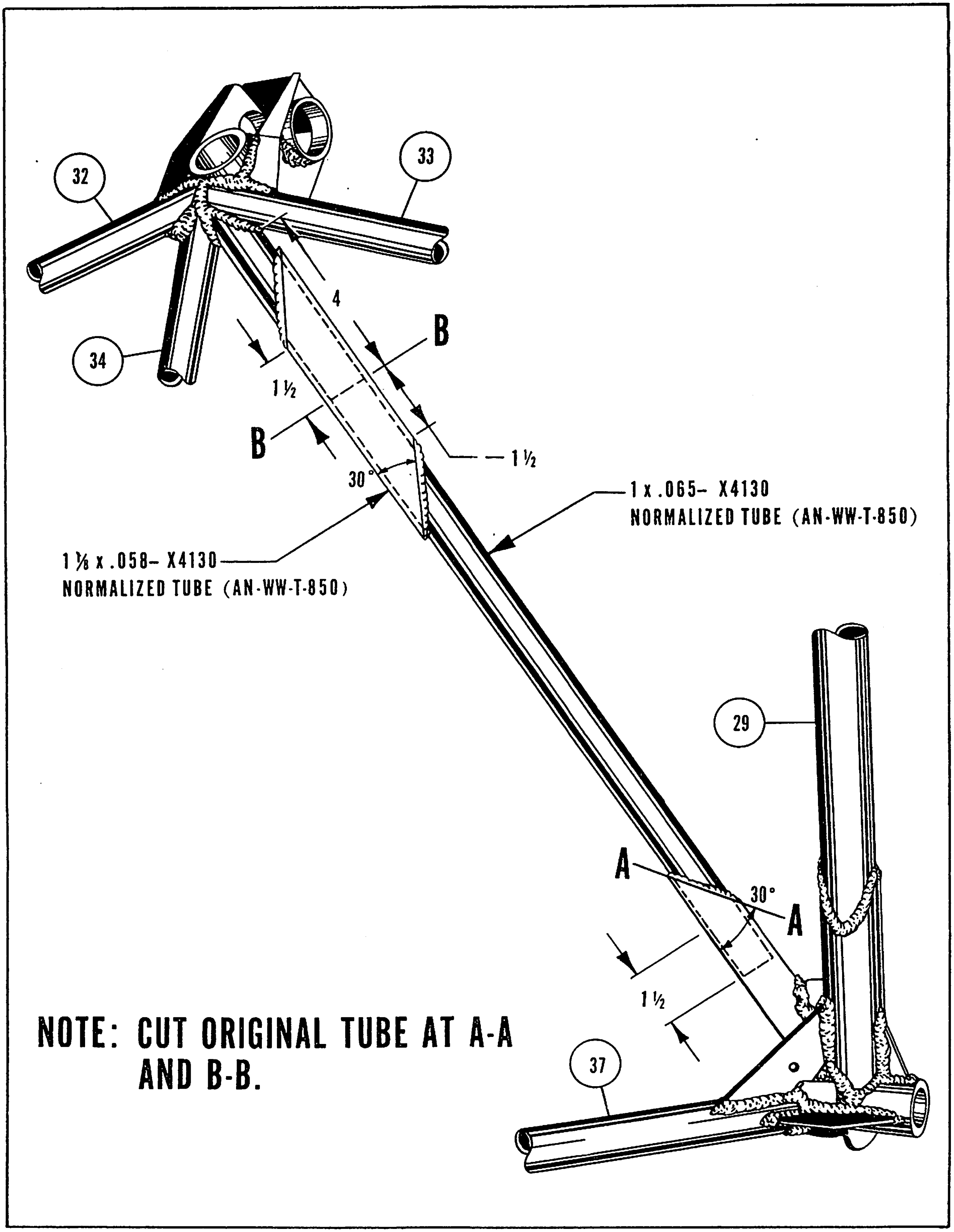


Figure 2-14. Splicing -34 and -35 Tubes

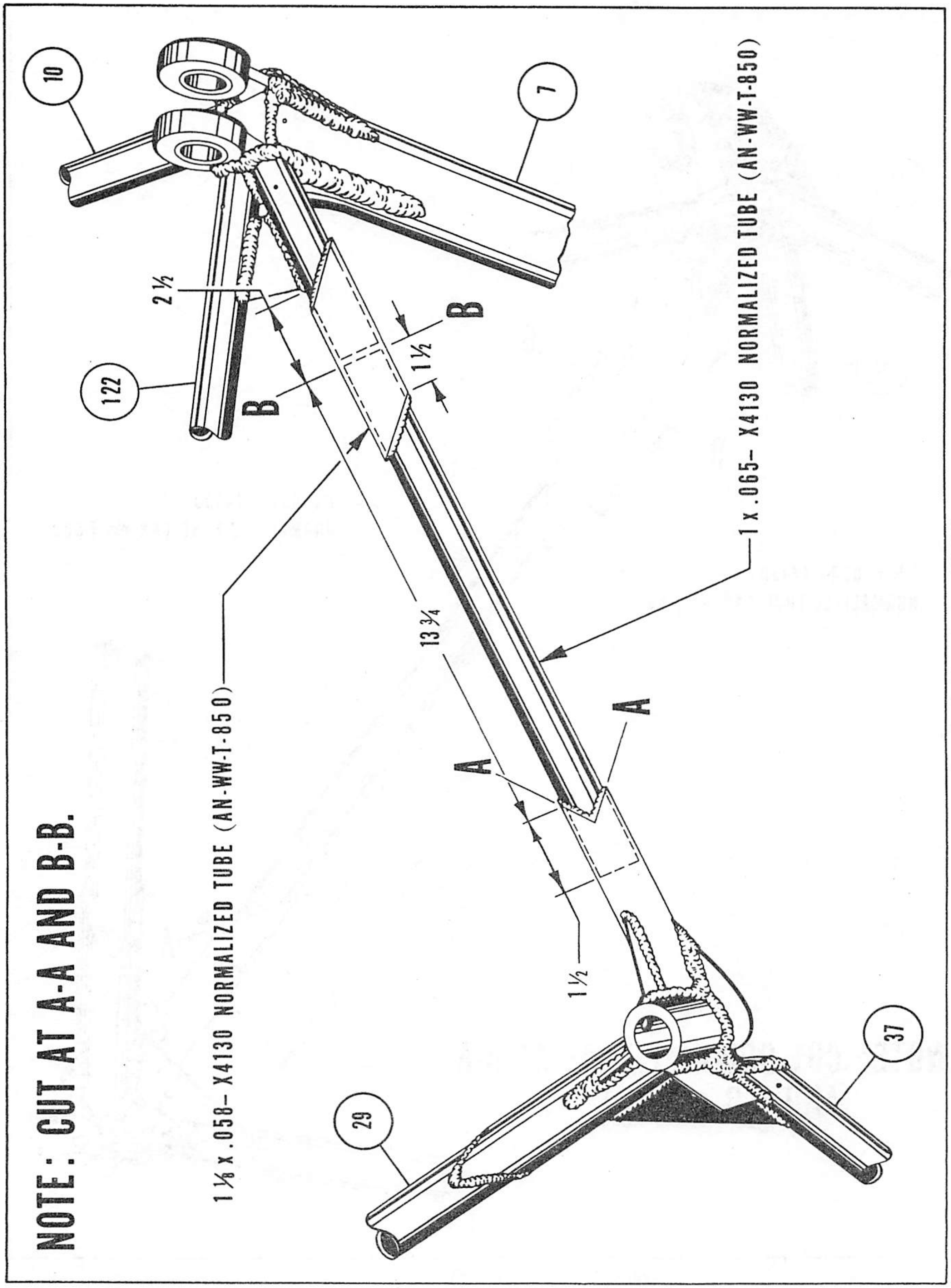


Figure 2-15. Splicing -28 Tube

2-80. REPLACING OUTBOARD CENTER SECTION TRUSS ASSEMBLY.

a. To accomplish this repair, it will be necessary to either drain and purge all fuel tanks or drain and remove all fuel tanks.

b. Remove batteries.

c. Drill out rivets and remove bottom leading edge of inboard wing. Install jack pads and place aircraft on jacks capable of supporting it in level position.

d. Remove propellers, cowling, right and left engine.

e. Disconnect all wiring, plumbing, and cables from the wing that is to be removed.

f. Remove wing flap and outboard wing. Place adjustable support under the outboard end of the opposite wing.

g. Remove main landing gear and landing gear doors on the side to be repaired.

h. Remove firewall attaching clamps and oil tank cover.

i. Remove all engine controls, plumbing, and electrical wiring from firewall and leading edge wing section forward of truss. Cap all exposed plumbing.

j. Drill out rivets and remove lower inboard and outboard nacelle skin and firewall. (Oil tank need not be removed.) See figure 2-16.

k. Remove inboard upper leading edge skin and battery stand.

l. Remove Nose Rib 2 and electrical junction box. Secure engine controls and wiring away from the field of operation.

NOTE

Remove liquidometer wires from Nose Rib 2 electrical junction box before removing the junction box from the aircraft.

m. Remove flap shaft and nacelle gearbox.

n. Remove gear shaft, chain, and lower chain sprocket and associated parts from the nacelle and truss assembly.

o. Remove front and rear fuel tanks, rear fuel tank fuel line, and front fuel tanks compartment on side to be repaired.

p. Remove magnetic compass and pilot's compartment upholstery on side to be repaired.

q. Remove cabin center aisle floor board, chairs, and main floorboard on the side to be repaired.

r. Remove front cabin window frame and glass. Remove upholstery between Bulkheads 5 and 6.

s. Cut out a section of the fuselage skin as shown in figure 2-17.

t. Remove sufficient skin and cut away a portion of Bulkhead 5.

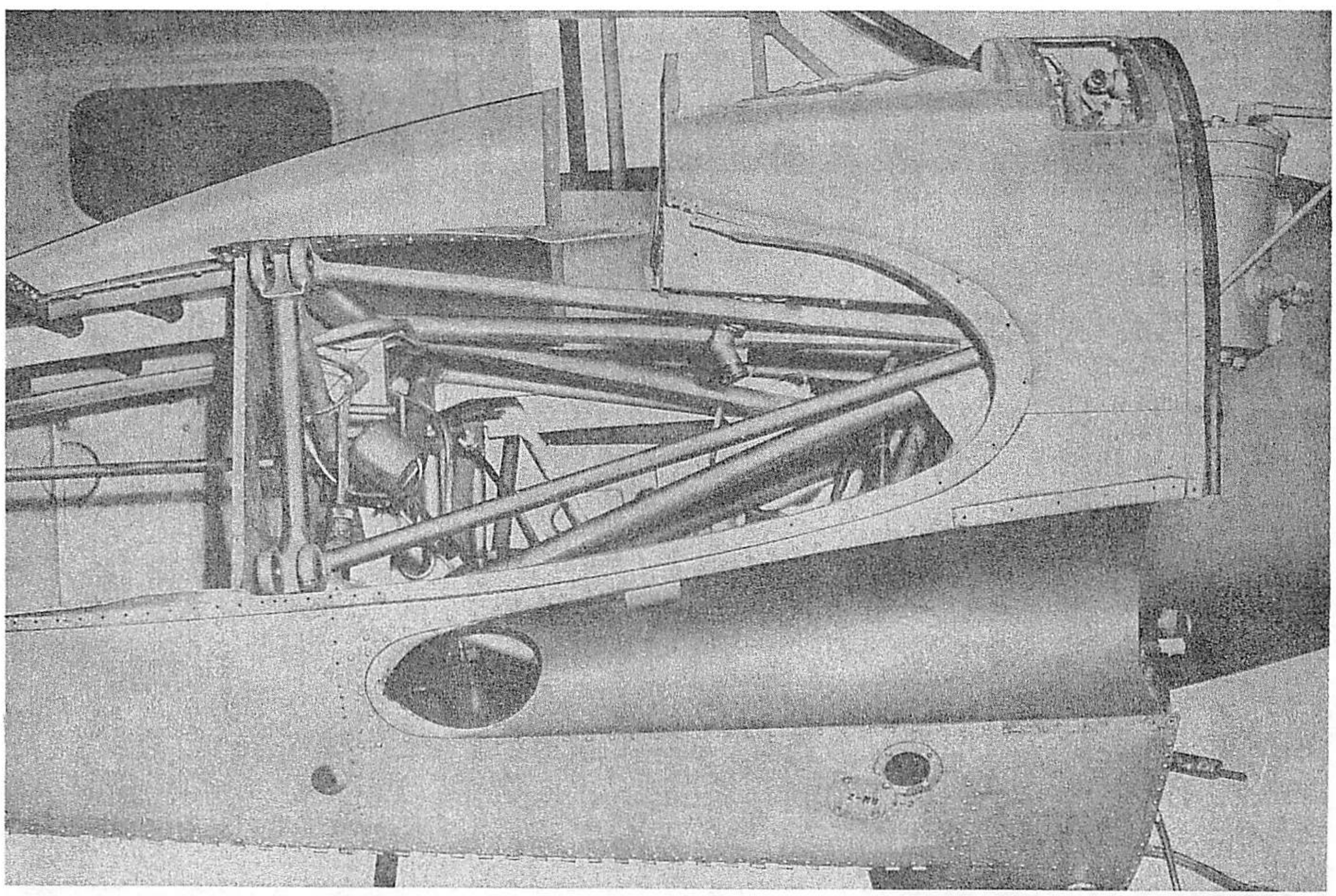


Figure 2-16. Removal of Nacelle Skin and Firewall

u. Place support under Bulkhead 6 with aircraft in level position. Remove jack from side of aircraft to be repaired.

v. With the aircraft in level position, install suitable tooling or jig to establish and hold reference points, thereby enabling the new truss member to be installed identically to the original installation. For best results, suspend the plumb-bob in a can of SAE 30 motor oil. See figure 2-18.

w. Check jig, or tooling for final dimensions and alignment. Provide support for the inboard wing section and drill out the rivets in the truss gussets that secure the truss to the remaining aft structure of Nose Rib 2.

x. Saw through truss drag tube four inches aft of splice weld.

y. Place support beneath the section of truss to be removed.

z. Support the section of truss to be removed and saw off both main truss members as close to the cabin fittings as possible.

aa. Grind off remaining portions of the outboard truss members and plate, flush with the cabin fittings.

ab. Grind off the weld and the outboard truss section gussets from upper and lower center section elliptical tubes. See figure 2-17. Drill six small holes (drilling first with a No. 40 drill, then following up with a No. 30 drill or a No. 30 reamer) along the upper and lower outboard truss attaching gussets parallel to the attaching gussets. This will allow a saw blade to be started above and below the outboard attaching gussets. Saw the attaching gussets loose from the center section of the center section truss, and remove plates from center section elliptical tubes.

ac. Remove slide tube, and all clamps and fittings from the truss section just removed, and install all parts on the new truss assembly.

ad. Fabricate a splice section of tubing from 1-1/2 inch diameter chrome molybdenum steel tubing, .049 wall thickness, approximately 11 inches long. Cut both ends of this section of tubing on the same angle as the drag tube fitting on the new truss assembly. See figure 2-19. Slide this splice section of tubing over the original truss drag tube where it will temporarily remain.

ae. Cut a section from 1-3/8 inch diameter chrome molybdenum steel tubing, .065 wall thickness, and approximately 9 inches in length. Insert one end of this section of tubing into the truss drag tube fitting on the new truss assembly. (When the new truss is in place this tube will butt against the original truss drag tube.) The actual correct length of this tube will be determined when the new truss is in place and may vary slightly from the specified 9 inches.

af. Install the straight edge between the upper and lower wing fittings on the new truss assembly. Locate the new truss assembly in place determined by the jig or tooling, and support the new truss assembly in this position. See figure 2-18.

ag. Tack weld the upper and lower main truss members both fore and aft. Recheck truss assembly for alignment.

ah. With the truss in proper alignment, install vertical brace between upper and lower cabin fitting while welding is being done. Start welding the outboard truss section gussets to the center section elliptical tubes, welding alternately from one side to the other. Constantly check the alignment of the outboard truss section to the jig in order to hold shrinkage and distortion to a minimum. See figure 2-20.

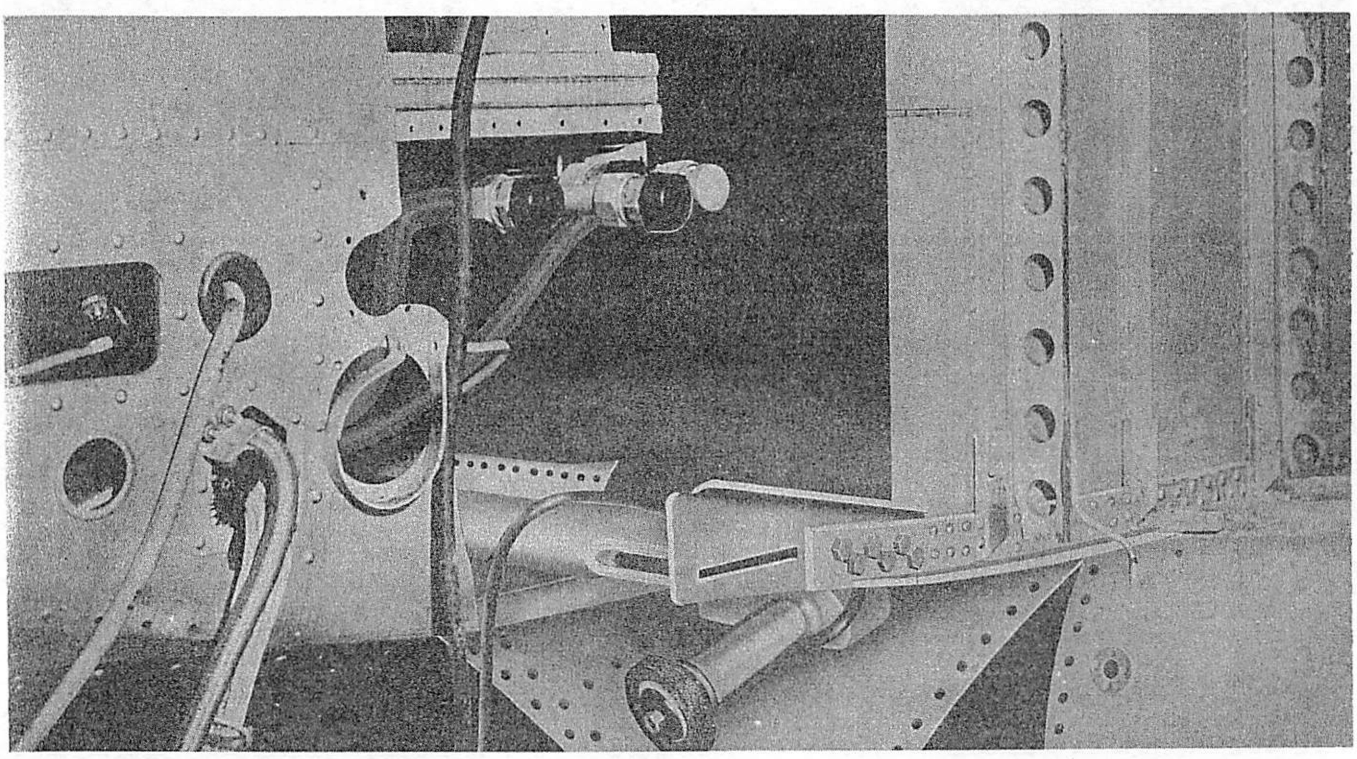


Figure 2-17. Grinding Outboard Truss Members

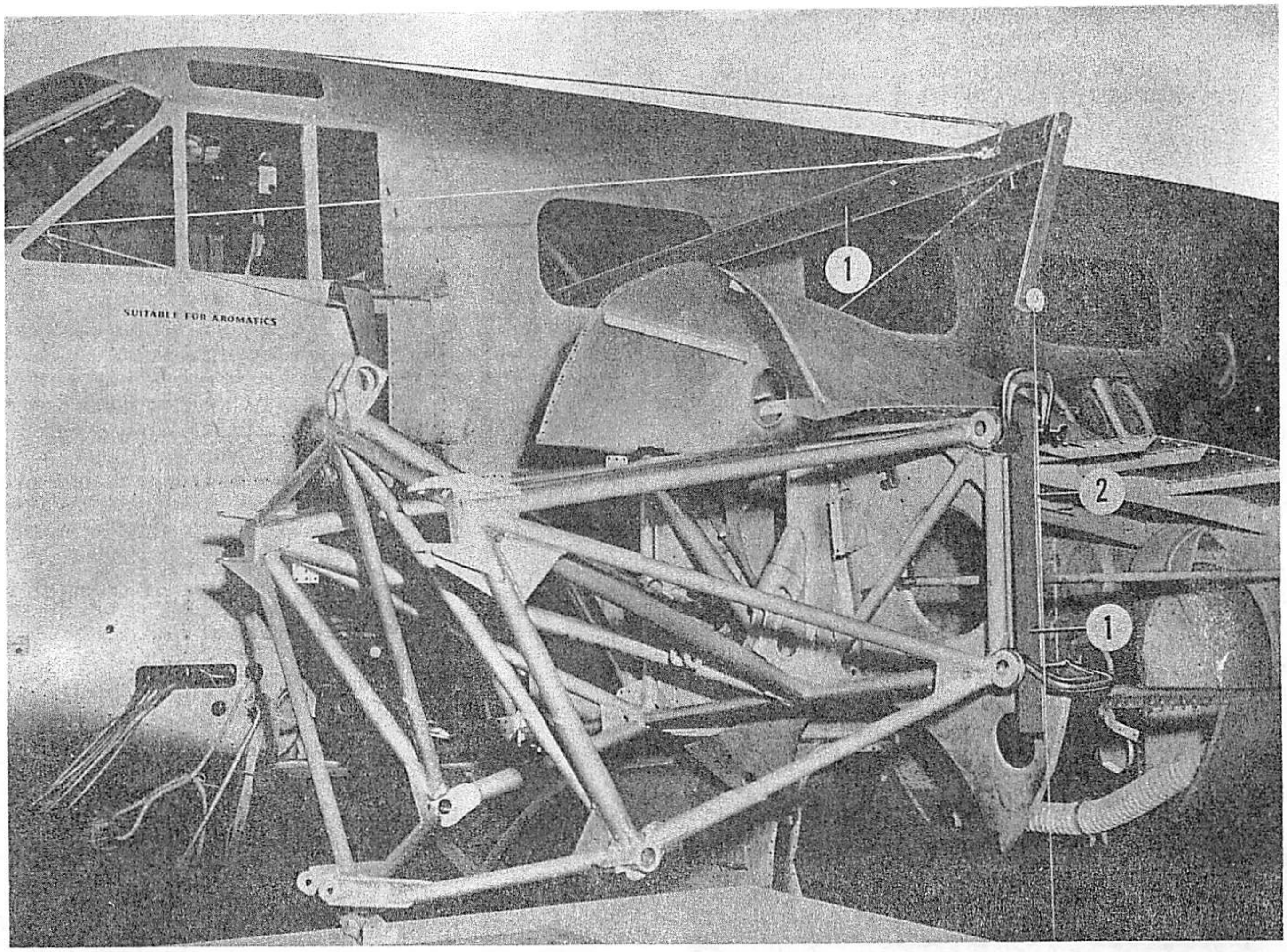


Figure 2-18. Alignment Jig

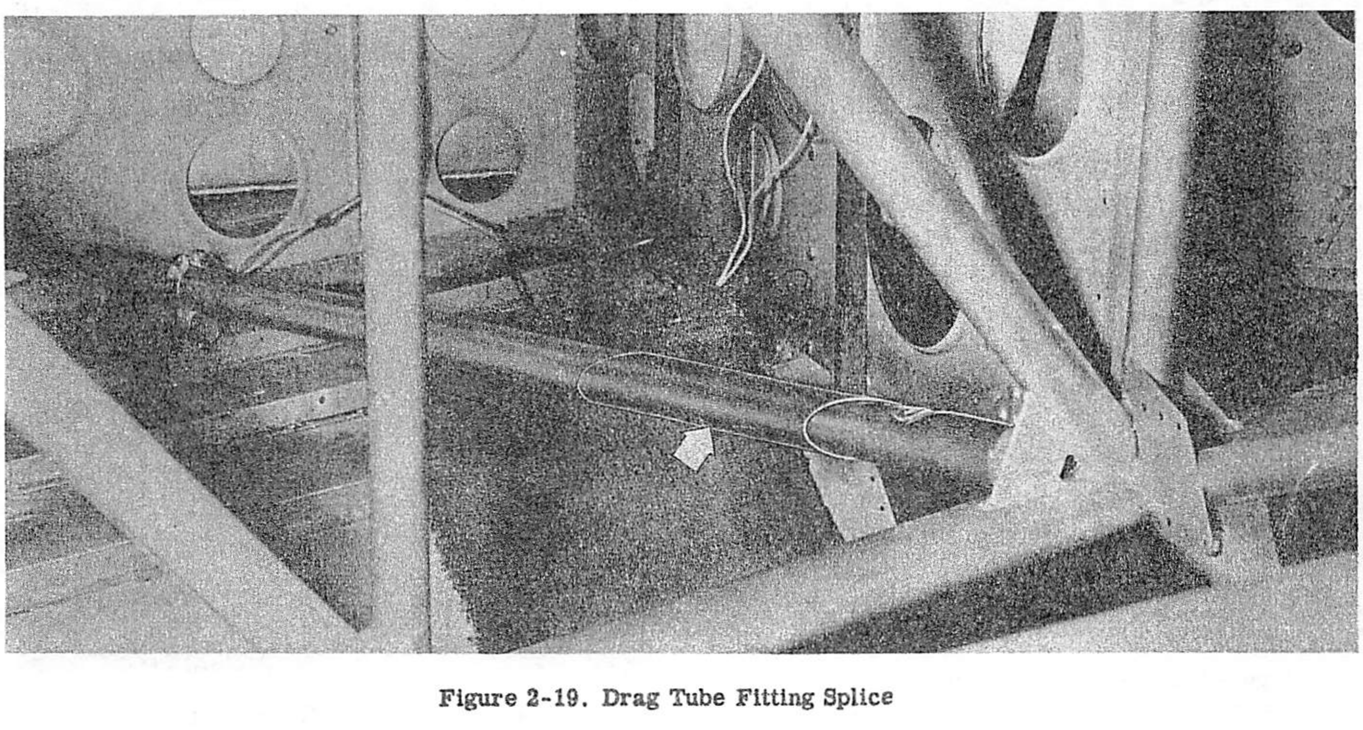


Figure 2-19. Drag Tube Fitting Splice

NOTE

It is essential that one man constantly check the alignment of the new truss assembly and jig, advising the welder when to alternate the welding.

After the outboard truss gussets have been welded to the center section elliptical tubes, weld the elliptical tubes to the cabin truss plate, terminating the weld 3/8-inch from the aft root rib angle. With the weld terminated at this point remove the bolts from the cabin truss plates, and spring the truss forward as far as possible, which will be from 1/8 to 3/16-inch. Place a sheet of asbestos between the cabin truss plate and the fuselage root rib angle. Pack the remainder of the weld area with web asbestos mud, and complete the weld.

aj. Slide the tube referred to in paragraph ae back against the original truss drag tube. Butt weld this tube to the original drag tube. Recheck the truss alignment and tack weld the forward end of this tube to the truss drag tube fitting.

ak. Grind off sufficient butt weld to allow the splice tube referred to in paragraph ad to slide forward over the butt weld and up to the truss drag tube fitting, allowing approximately 3/16-inch clearance between the two for welding. Weld both ends of the splice tube including the truss drag tube fitting in the forward weld, See figure 2-19.

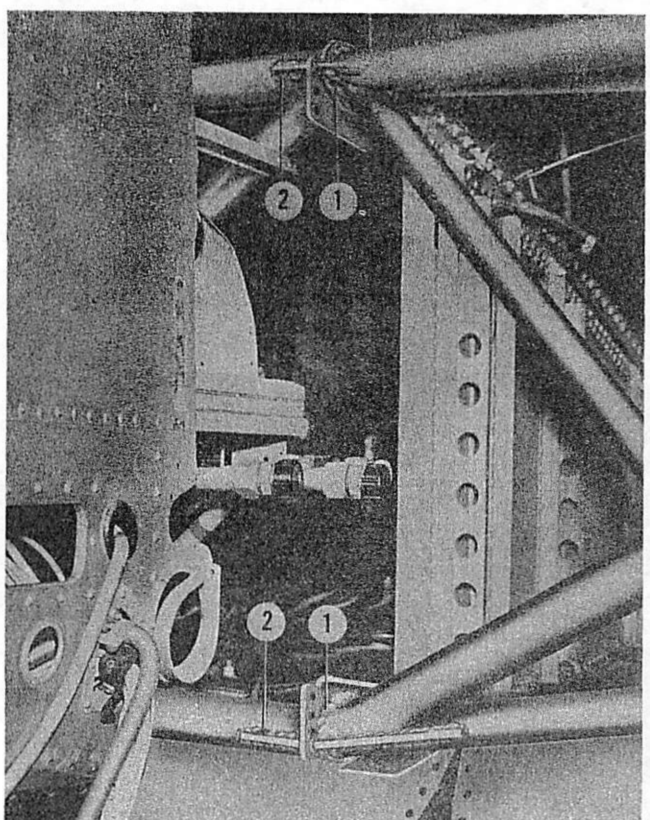


Figure 2-20. Alternate Welding Center Section to Outboard Section Gussets

al. Magnaflux all welds.

am. Lionoil all weldments and welded sections, clean with wire brush, prime and paint.

an. Reassemble, using the reverse of the above procedure, except as follows:

1. Splice the root rib skin just aft of the truss cabin fittings using .051, 24STAL sheet approximately 14 x 24 inches. Use two rows of rivets 3/4-inch apart, 3/4-inch rivet spacing, adding 1/2 x 7/8 x .051 bulb angle reinforcement. Use AN470AD4 rivets of sufficient length. See figure 2-21.
2. Splice stringers under cabin floorboards with a standard fuselage butt splice, using AN470AD4 rivets; eight rivets installed in each end of the splice.
3. Splice new section on lower end of Bulkhead 5 using a section of .040, 24STAL sheet made up to fit inside of Bulkhead 5, and approximately 4 inches long, using AN470AD4 rivets. See figure 2-22.
4. Splice the upright stringer above the main spar at Bulkhead 5 with 5/8 x 1 inch, .065, 24STAL angle clip, approximately 6 inches long. Pick up rivets through the web using AN470AD4 rivets. See figure 2-23.

CAUTION

Reassembly all of the leading edge skin before installing lower nacelle and firewall assembly.

NOTE

When ordering an outboard center section truss assembly, it is necessary to state the type of aircraft, the serial number, and specify right or left.

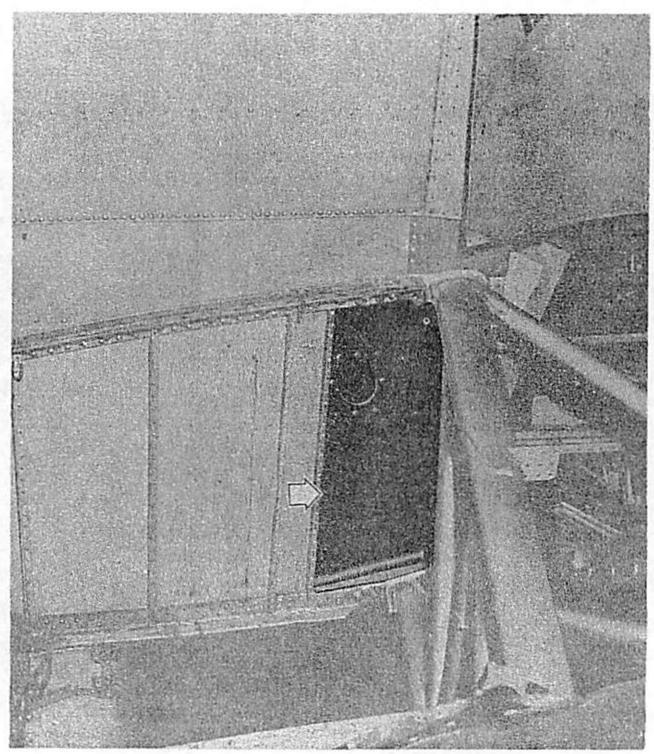


Figure 2-21. Root Rib Skin Splice

2-81. FUEL TANKS.

2-82. To repair cracks or breaks in a fuel tank, proceed as follows:

a. Immerse tank completely in a hot (82°-88°C) (180°-190°F) cleaner solution made from a concentration of Oakite No. 64 (8-10 ounces per gallon), Kelite No. 60 (4 ounces per gallon) or Turco 3266 (6-7 ounces per gallon).

NOTE

Excessively greasy or oily tanks should be rinsed inside and out with a suitable solvent (trichlorethylene or solvent, Specification P-S-661A) before immersion in the hot tank cleaner.

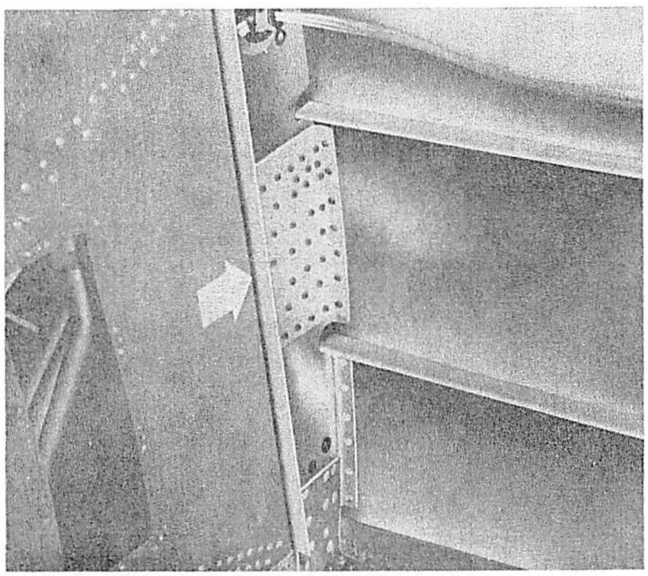


Figure 2-22. Lower Splice, Bulkhead 5

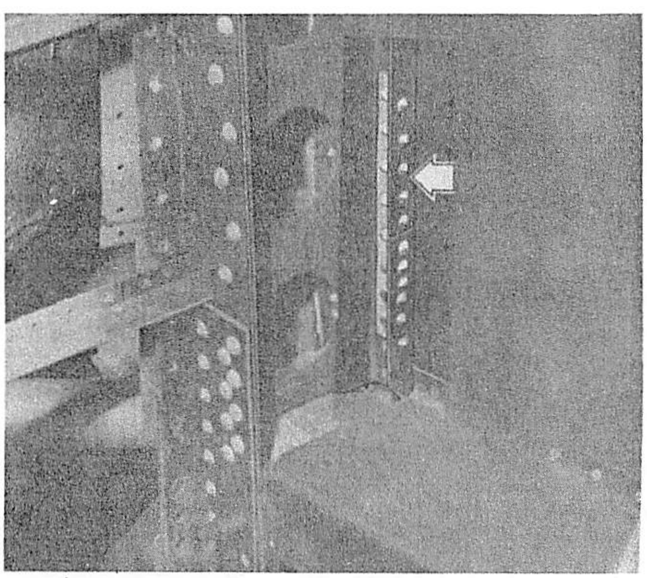


Figure 2-23. Upright Splice, Bulkhead 5

b. Leave immersed as necessary to clean, up to a maximum of four hours. Slightly agitate the solution within the tank to facilitate faster removal of the flushing film.

c. When inspection shows all film has been dissipated, drain solution from tank, wash and rinse thoroughly with cold running water.

d. Acidize by immersing in a water solution of Oakite 84A (8-10 ounces per gallon) for 5 to 10 minutes at room temperature (21°-32°C) (70°-90°F).

e. Drain tank for a few minutes, then wash thoroughly with clean cold water.

f. Follow with a dip in clean hot water to facilitate drying. Use air blast to remove last traces of water from between joining surfaces. Tank is now ready for repair.

2-83. After repair or alternations have been made on the tank by welding, remove welding flux as follows:

NOTE

Welding on fuel tanks is accomplished by using hydrogen gas and silicon (hard) welding rod.

a. Immerse in a solution of Oakite No. 84A (8 to 10 ounces per gallon) at room temperature (21°-32°C) (70°-90°F), for 15 to 30 minutes (or until all flux has been dissolved).

b. Rinse with clean, cold running water before neutralizing. Thorough rinsing is essential.

c. Neutralize by immersing tank in a hot solution (82°-88°C) (180°-190°F) of Oakite No. 61 (8-10 ounces per gallon) for 15 to 30 minutes.

d. Rinse thoroughly in cold running water.

e. Rinse in hot water to facilitate drying. Dry, using dry, compressed air.

f. Seal all openings and air test tank at 5 psi.

2-84. (Deleted).

2-85. For refinishing the exterior of the fuel tank proceed as follows:

a. Remove any paint left on the tank and sandpaper any rough spots, rinse and permit to dry.

b. Prime with one coat of zinc chromate primer.

c. Spray with two coats of aluminized lacquer (MIL-L-6806).

SECTION III

TAIL GROUP

TABLE OF CONTENTS

3-1. TAIL GROUP.

3-2. DESCRIPTION.

3-3. The horizontal stabilizer, elevator, two vertical stabilizers and two rudders compose the empennage parts. Two trim tabs are attached to the elevator and one to the left rudder. Angle of incidence of horizontal stabilizer to centerline of the fuselage is negative two degrees. See figure 3-1.

3-4. HORIZONTAL STABILIZER.

3-5. The horizontal stabilizer is a stressed-skin panel of all aluminum-alloy construction (figure 3-2). The ribs are attached to front and rear spars and are supported along this span by extruded angle stringers. The stabilizer is attached to the fuselage in fixed alignment by attaching brackets bolted to the fuselage and fillister head screws through the support angles on Fuselage Bulkheads 13 and 15.

3-6. VERTICAL STABILIZER.

3-7. Two vertical stabilizers are attached to the outer ends of the horizontal stabilizer by means of fillister head screws through angles riveted to the horizontal stabilizer covering. They are stressed-skin panels of aluminum-alloy construction (figure 3-2). The ribs are attached to front and rear spars and are supported along this span by stringers made from .032 sheet material (see figure 3-1).

3-8. ELEVATOR.

3-9. The elevator is a fabric-covered, aluminum-alloy structure, similar in construction to other control surfaces. It is hinged to the horizontal stabilizer at five points on prelubricated bearings mounted in cast aluminum-alloy brackets which are bolted to the horizontal stabilizer. The elevator is shaped by 18 ribs attached to a spar. The trailing edge is made of Alcoa Section K-1508 and the nose is of a metal skin supported by 22 nose ribs. Four balancing weights, two on each side, are fastened to the inside of the nose skin to give static and dynamic balance. Two trim tabs, hinged to bulkheads attached to the ribs, are fitted into the trailing edge of the elevator, one on each side of the centerline.

3-10. RUDDER.

3-11. A rudder is attached to the rear spar of each vertical stabilizer. The rudders are fabric-covered control surfaces of similar construction to the ailerons, flaps, and elevator. Each rudder is shaped by seven

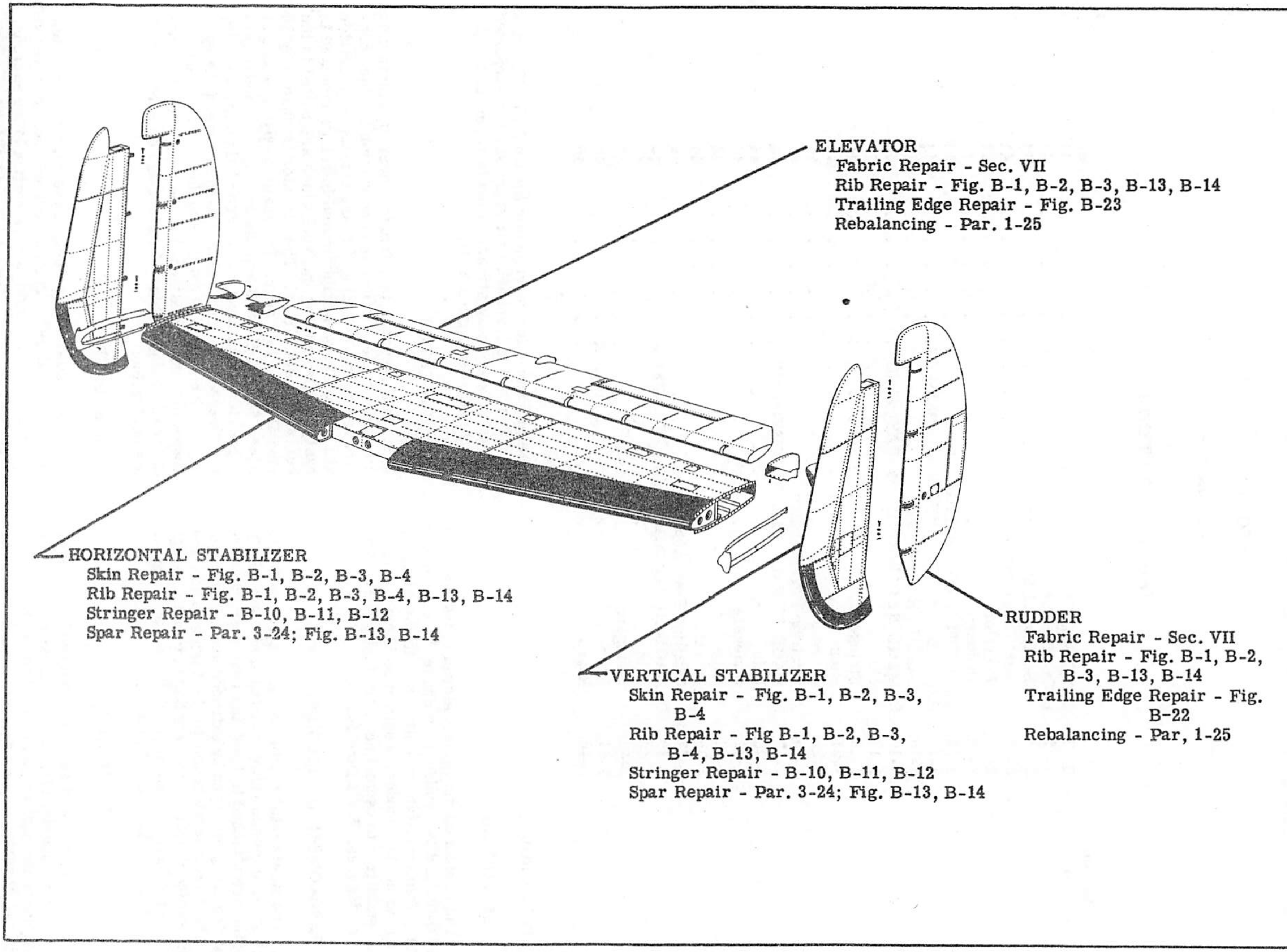


Figure 3-1. Tail Group Exploded

ribs fastened to a spar and reinforced by diagonal braces and bulkheads. The nose section of the rudder is made of metal skin supported by ten nose ribs, and the trailing edge is made of Alcoa Streamline Tube Section No. T-535. A balancing weight is built into the upper front part of each rudder to give static and dynamic balance. A trim tab is built into the trailing edge of the left rudder only. Each rudder is hinged on four prelubricated bearings mounted in cast aluminum-alloy brackets which are bolted to the rear spars of the vertical stabilizers.

3-12. DEFINITION OF DAMAGE.

3-13. Damage may be defined as follows:

a. Negligible damage is damage that will not affect the airworthiness of the tail group and does not require immediate attention.

b. Damage repairable by patching is damage that may be repaired by covering or reinforcing without removal other than trimming of a portion of the skin or structure.

c. Damage repairable by insertion is damage requiring the removal and replacement of a portion of the tail group skin or structure.

d. Damage repairable by replacement is damage unrepairable by patching or insertion, but which may be repaired by installing a new part. Damage requiring replacement, but which cannot be replaced because of structural design, will necessitate replacement of the entire damaged assembly.

3-14. PROCEDURE FOR REPAIR OF DAMAGE.

3-15. NEGLIGIBLE DAMAGE. Damage of this classification shall be limited for the entire tail group to surface dents and scratches in the skin. The dents must not substantially change the contour of the airfoil and must be carefully investigated for indications of structural damage.

3-16. DAMAGE REPAIRABLE BY PATCHING.

3-17. SKIN.

3-18. Small holes, cracks, or breaks in the metal skin covering of the horizontal or vertical stabilizers may be patched, with the exception of those occurring in the leading edge skin. The leading edge skin must maintain a strictly smooth contour, necessitating a flush type patch. It would be extremely difficult to form a patch to fit this contour properly; therefore, repair by insertion is recommended. Other sections of the stabilizers may be repaired with surface patches of the following types:

a. Round holes up to one inch in diameter may be reinforced by a washer installed as shown in figure B-4. Reinforcing, rather than covering the hole, permits the use of a bucking bar in setting the rivets. If Cherry rivets are used, a disc may be substituted for the washer.

b. Small breaks or punctures in the skin may be repaired as shown in figures B-2 and B-3. The ragged edges should be cut away as shown by the dotted lines, so that no sharp corners remain. To calculate the minimum number of rivets to be used, multiply the length of the hole after trimming by eight. As an example, assume the length of the hole is 2.5 inches: 8 x 2.5 = 20 rivets. Therefore, a minimum of 20 rivets must be used on each side of the hole, or 40 rivets for the entire patch.

c. To repair cracks in the skin, first drill stop-holes at each end or sharp corner of the crack (see figure B-1). The patch plate should be cut large enough to clear

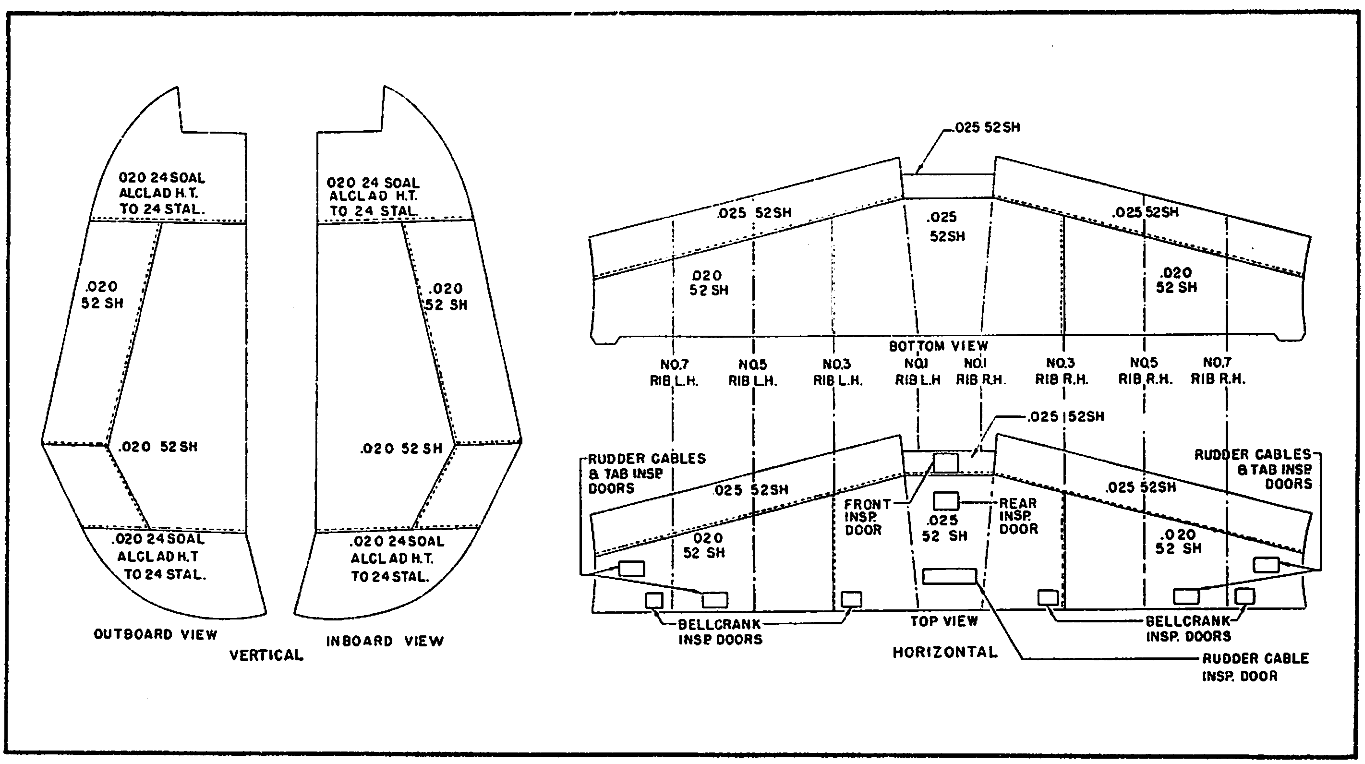


Figure 3-2. Empennage Skin Plating Diagram

all parts of the crack by 7/8-inch. Cherry rivets, if available, may be substituted for the standard rivets.

3-19. STRINGERS.

3-20. Cracks in stringers of the tail group may be repaired by reinforcing the damaged member with an extruded section of the hand-formed splice angle, as follows:

a. An extrusion of the same cross section as the original stringer may be used as patch material. The reinforcement must fit smoothly into the stringer. In some cases, it is necessary to remove the bulb and radius corners of the patch extrusion (see figure B-10).

b. The hand-formed splice angle should be made of .064, 24ST sheet and attached as shown in figure B-11. If the flange on the extrusion or stringer to be repaired is under 3/4-inch, the radius of bend for the splice angle should be 3/16-inch. The splice angle should be formed on a brake machine if one is available.

3-21. RIBS.

3-22. Damage to the structure of all ribs of the tail group can be repaired as follows:

a. Small cracks or breaks, entirely within the rib web, may be repaired by methods described for skin patches, paragraph 3-18c, if space permits.

b. Damage to the formed flanges extending down into the web or in ribs that are too small for a skin-type patch, will require a patch formed to fit the flange of the rib (see figure B-13 and B-14). The crack or break must be covered with a patch which will extend at least 3/4-inch each way from the edge of the crack. Stop holes should be drilled at the ends of cracks and breaks should be properly trimmed to prevent spreading. In riveting the rib patch to the rib flange, the skin rivets may be used with an additional rivet between the original skin rivets. Rivets must not be spaced closer than 3/8-inch.

3-23. SPARS.

3-24. All cracks or breaks in spar members shall be reinforced with a patchplate extending across the full cross section of the spar. The patch should be flanged to fit inside the spar flanges and should be attached to the spar flanges, using the original rivet holes in the spar and skin and spacing new rivets midway between the old rivet holes. A double row of rivets, staggered with 1/2-inch spacing, should extend the full length of each side of the patch across the face of the spar. If the patch falls on a lightening hole, the patch should be formed to fit the lightening hole flanges. Patches of this type are more easily formed from 24SOAL Alclad, heat treated after forming to 24STAL Alclad. Repair material must be next greater gage than the spar to be repaired.

3-25. TRAILING EDGE REPAIRS.

3-26. Cracks or small breaks between rib members may be repaired by a reinforcing patch as shown in figures B-22 and B-23. Care must be exercised in inserting the splice block to avoid further damage, especially in the case of the rudder trailing edge which is formed of streamlined tubing. A minimum of three rivets must be used on each side of the break (see figures B-22 and B-23).

3-27. DAMAGE REPAIRABLE BY INSERTION.

3-28. SKIN.

3-29. When damage to the skin of the tail group is too extensive to be repaired by patching, the skin should be cut back to the surrounding stringers and ribs, and a new piece of skin inserted. The new piece of skin should be the next greater thickness than the damaged piece and of the same material. A minimum spacing of 3/8-inch between rivets must be maintained.

3-30. Skin joints are classified as either longitudinal or transverse.

a. The longitudinal skin joints lie along stringers and spars, with the skin attached as shown in figure B-9. Skin joints are made at points of lesser load wherever possible; thus a new joint at the site of an old joint may use the same rivet spacing. However, a skin joint where no joint existed previously should be made with twice the number of rivets which attached the skin to the frame at that point, since that area may be more heavily loaded.

b. The transverse skin joints lie along the ribs and skin is attached as shown in figure B-8. The original rivet spacing may be used if the joint is made at an original skin joint; otherwise, the replacement skin should be cut to lap over the rib by 1/2-inch, riveted with a staggered row of rivets, and with a spacing equal to the original rivet spacing in the rib.

3-31. STRINGERS.

3-32. Damage requiring removal of a portion of a stringer may be repaired by cutting out the damaged portion and inserting a new extruded section. This section may be held in place with two splices, installed as in paragraph 2-42 (b). See figures B-11 and B-12.

3-33. RIBS.

3-34. Serious damage to a rib may require replacement of a portion of the rib. Remove the damaged part by making a straight cut across the rib. Form a new portion the same size as the damaged portion which was removed, using material of the next greater thickness and the same alloy. This new part may be spliced to the rib by using the same procedure as that described in paragraph 2-44. See figure B-13 and B-14. There should be a staggered double row of rivets on each side of each joint.

3-35. SPARS.

3-36. Damage to the spars of the tail group may be repaired by cutting away the damaged portion and forming a new section to replace it. This repair may be made by following the procedure outlined for the repair of the ribs in paragraph 3-34 above.

3-37. TRAILING EDGE REPAIRS.

3-38. To replace a damaged trailing edge of the elevator or rudders, cut out the damaged section and replace it with a new piece. The new section should be spliced in as shown in figures B-22 and B-23. A splice should never be made at a rib, due to the difficulty of installing a splice block directly at a rib. It is usually most convenient to make extremities of the cut center between the trailing edge ribs. The splice block may be made of 52SH or any stronger aluminum alloy. A minimum of three rivets on each side of the cut will be necessary

3-39. DAMAGE REPAIRABLE BY REPLACEMENT.

3-40. SKIN.

3-41. If more than 50 percent of a sheet of skin is damaged, the entire sheet should be replaced. Use material of same thickness and equal strength.

3-42. STRINGERS.

3-43. If damage to a stringer is so extensive that repair by insertion or patching is impractical, the entire stringer should be removed and replaced. In some cases, this may necessitate the removal of skin in the area of the damaged stringer.

3-44. RIBS.

3-45. Ribs may be replaced as follows:

a. To remove ribs in the stabilizers, it will be necessary to remove the top skin in the area of the damaged rib. This should cause no undue inconvenience, since ordinarily rib damage requiring replacement will involve the surrounding skin so that extensive patching or replacement will be necessary. If 50 percent of a rib is damaged, the entire rib should be replaced.

b. Elevator and rudder ribs may be replaced by slitting the fabric in the vicinity of the damage and pulling it back away from the damaged ribs. If 50 percent of the rib is damaged, the rib should be replaced. Form the new rib of the same material as the old rib.

3-46. SPARS.

3-47. Spars may be replaced as follows:

a. The stabilizer spars are built in sections and one or more damaged section may be replaced; however, the repair will be very difficult. If it is necessary to replace a stabilizer spar, the complete stabilizer assembly should be replaced.

b. Extensive damage to elevator and rudder spars will necessitate replacement of the entire assembly.

SECTION IV

BODY GROUP

TABLE OF CONTENTS

4-1. BODY GROUP.

4-2. DESCRIPTION.

4-3. The fuselage is a semi-monocoque, all-metal structure with stressed metal skin covering. The structure (See figure 4-1) consists of stringers running fore and aft through the fuselage and attached to the bulkheads. The flooring in cabin and pilots' compartment is supported by a structure of intersecting channels attached to the bulkheads and skin. The skin covering is riveted to the stringers and bulkheads (See figure 4-2).

4-4. DEFINITION OF DAMAGE.

4-5. Damage may be defined as follows:

a. Negligible damage is damage that will not affect the airworthiness of the fuselage structure and does not require immediate attention.

b. Damage repairable by patching is damage that may be repaired by covering or reinforcing a portion of the fuselage skin or structure.

c. Damage repairable by insertion is damage requiring the removal and replacement of a portion of the fuselage skin or structure.

d. Damage repairable by replacement is damage unrepairable (or too extensive to repair) by patching or insertion but that may be repaired by replacing the part.

4-6. PROCEDURE FOR REPAIR OF DAMAGE.

4-7. NEGLIGIBLE DAMAGE.

4-8. Negligible damage to the aircraft frequently may be caused by flying rocks, missiles, accidental abrasion, or inadvertent actions of personnel. Shallow dents, small, clean-edged holes, minor scratches, and bruises may be considered as negligible when not occurring in highly stressed portions of webs, angles, or skin. Cracks and tears must be checked or corrected to prevent spreading. A 3/32 to 1/8-inch hole, drilled at the end of a crack, is a temporary

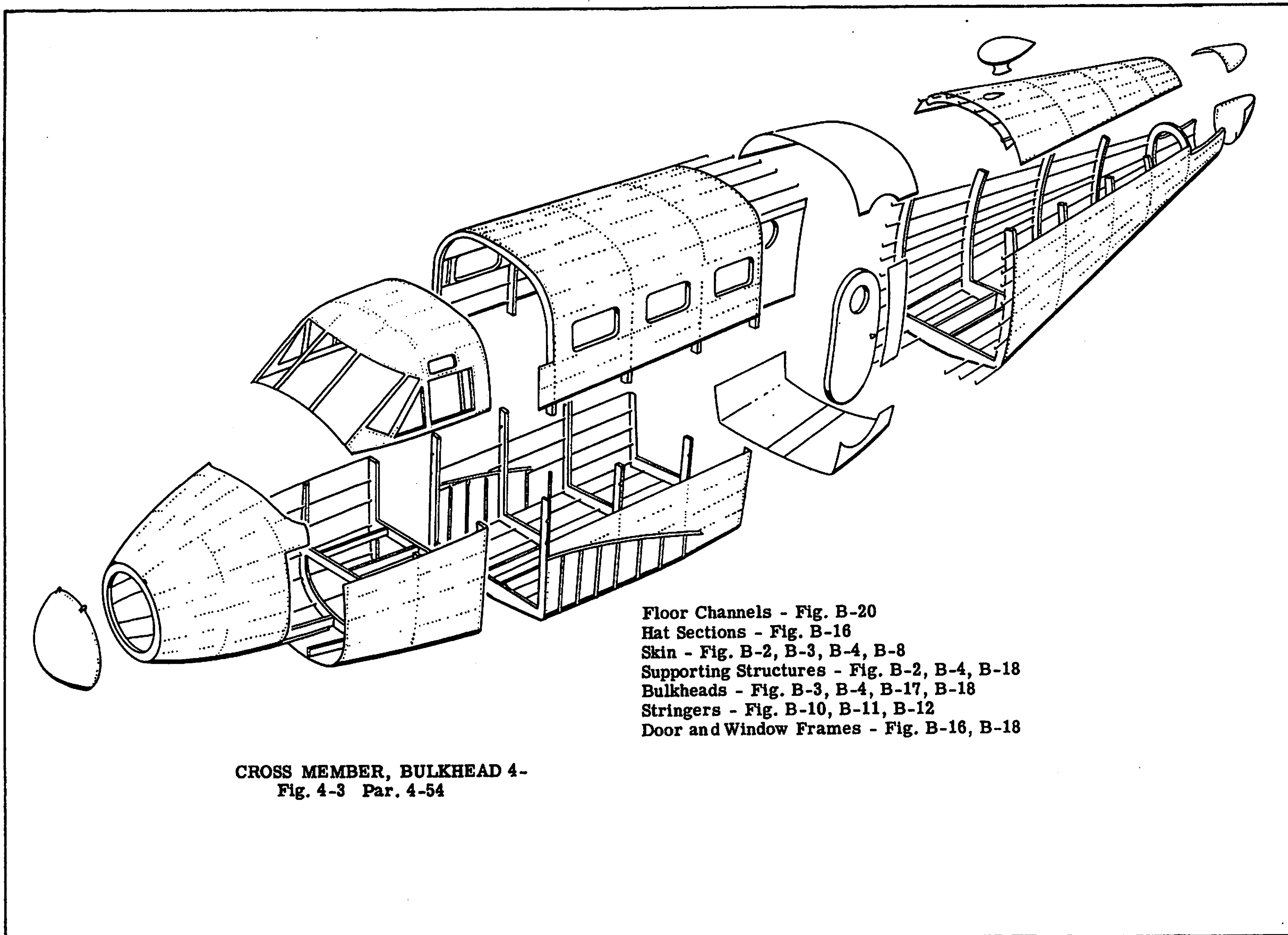


Figure 4-1. Fuselage Exploded

means of checking its growth. Care must be taken that the hole is actually drilled in the end of the crack, since an improperly checked crack may continue to grow. Web damage in bulkheads must be carefully considered, as the same injury which could be accounted negligible in one locality may be detrimental in another. Dents or holes in the hydro-pressed parts should not be allowed to remain; however, dents or holes in thin sheet sections of some of the bulkheads are not critical, and fairly large damaged areas of such sections may be considered negligible.

4-9. FLOOR CHANNELS.

4-10. Floor channels have quite heavy loads imposed upon them by the cabin seats and cargo, and care must be taken that they develop maximum strength. Small holes will be allowed in the webs provided they have ample edge distance and the edges are smoothed so that there is no chance for cracks to elongate; however, holes or cracks in flanges will not be allowed, and must be repaired. Slight bends of fairly large radii may be straightened. Any damage to the floor supports adjacent to the control column, torque tube, or landing gear and flap motors must be carefully considered since distortion may cause faulty operation.

4-11. HAT SECTIONS.

4-12. The hat sections are quite heavily stressed and any damage to them must be carefully considered, especially in the formed portions. Small holes, having ample edge distance and which do not occur in the radius of a bend and which do not have any rough corners, may be disregarded if the radius is large, but holes or dents in the formed portion will require repair.

4-13. SKIN.

4-14. Small clean holes and small dents may be considered negligible damage to the skin. Jagged holes should be cleaned out and sharp dents smoothed out where possible, as sharp dents or jagged holes are likely to develop cracks which could spread. Stop-drill cracks with a 3/32-inch drill at the extreme end of the crack, to curtail growth. It is recommended that holes be patched as soon as possible to avoid annoying airstream whistles. Although the damage to the skin may be considered negligible from a structural standpoint, it may destroy the outer corrosion-protective coating.

4-15. SUPPORTING STRUCTURES.

4-16. Small dents in the radio rack will be considered negligible; however, since they are easily repaired, dents should be smoothed out as soon as possible. Small, clean holes will also be considered negligible if they do not occur in a flange and have ample edge distance.

4-17. DOOR AND WINDOW FRAMES.

4-18. The door and window frames are quite heavily stressed parts. Small holes in the cabin door and baggage door frame may be considered negligible if they occur in the web and have ample edge distance.

4-19. DAMAGE REPAIRABLY BY PATCHING.

4-20. BULKHEADS.

4-21. Cracks in bulkheads or holes which occur near the flanges of the formed sections should be repaired immediately as they weaken the structure considerably. Repair may be made by forming a patch of 24S aluminum-alloy material of the next gauge heavier than the part being repaired. It will be necessary, usually, to form the patch in the annealed or "O" condition, after which it will be necessary to heat-treat the patch to the "T" condition before installation on the part to be repaired. Refer to figure B-17 for details of the repair. Holes or cracks in the flat portions of the thin web of some of the bulkheads may be patched as shown in figures B-3 and B-4.

4-22. STRINGERS.

4-23. Patch stringers as follows:

a. Cracks in the stringers may be repaired by reinforcing the damaged member with an extruded section or a hand-formed splice angle.

b. An extrusion of the same cross section as the original stringer may be used as patch material. The reinforcement must fit smoothly into the stringer, even if it is necessary to file the bulb angle and corner radius of the patch extrusion. Refer to figures B-10 and B-11 for details of the patch.

c. In the event a hand-formed splice angle is used, the angle should be made of 24ST sheet, .064 inch thick. Radius of bend for this angle should be 3/16 inch.

d. If it is necessary to splice more than one stringer stagger the splices in such a manner that no two splices fall at the same station (see figure B-12).

4-24. FLOOR CHANNELS.

4-25. Cracks in floor channels may be patched by inserting a patch of the next gauge heavier material. If the crack or damaged portion occurs in the flange, it will be necessary to form the patch to fit the inside contour of the channel. 24ST material should be used for patch material (see figure B-12).

4-26. HAT SECTIONS.

4-27. Damage to hat sections may be repaired by the addition of a splice plate of 24ST material formed to fit over the top of the hat section. It will be necessary to form this patch in the annealed or "O" condition and heat-treat to the "T" condition before installation (see figure B-16).

4-28. SKIN.

4-29. Patch holes in skin as follows:

a. Various sizes of damaged areas in the skin will require different types of patches. Either a flush-type patch (see figure B-3) or the external type patch (see figure B-2) may be used.

b. Round holes up to one inch in diameter may be reinforced by a washer of the same thickness and material as the original (see figure B-4). In the event it is impossible to buck the rivets, some form of blind riveting device may be used.

c. Small breaks or punctures in the skin may be repaired as shown in figures B-2 and B-3. The ragged edges should be cut away as shown by the dotted lines, so that no sharp corners remain.

4-30. SUPPORTING STRUCTURES.

4-31. Cracks and holes in the supporting structures may be repaired by forming a patch to fit the contour of the supporting structure in the case of flanged channels (see figure B-18). Cracks or holes in flat sections of supporting structure may be repaired by inserting a reinforcing patch over the cracks or holes (see figures B-2 and B-4).

4-32. DOOR AND WINDOW FRAMES.

4-33. Damage to the hat sections, which constitute the cabin window frames, may be repaired by forming a sheet of the same material and of the next gauge thicker than the hat section being repaired, installing it as shown in figure B-16. The cabin door frame is made of a flanged section of sheet metal and is quite heavily stressed. The left-hand side of Bulkhead 8 acts as the front side of the cabin door frame. Patches may be made to the cabin door frame in the same manner as patches on bulkheads (see figure B-18).

4-34. DAMAGE REPAIRABLE BY INSERTION.

4-35. BULKHEADS.

4-36. Damage to bulkheads which requires insertion should be very carefully considered, as some of the bulkheads are built of relatively small sections. In many cases, although from a structural standpoint, there is no objection to splicing in an insertion, it will be much easier to replace a section. An insertion may be made to the bulkhead by trimming out the damaged area and inserting a formed section which duplicates the damaged portion removed. If a replacement or salvaged bulkhead is available, it will be more satisfactory to cut a section from the other bulkhead which duplicates the damaged portion removed. If a replacement insertion is not available, one may be formed from SO material and heat-treated to the ST condition after forming. Secure insert into bulkhead by means of a splice patch (see figure B-18).

4-37. STRINGERS.

4-38. Damage to stringers may be very easily repaired by trimming out the damaged area and inserting an extrusion of the same type to replace the trimmed away portion. The replacement insertion will be attached to the original stringer, as shown in figures B-10 and B-11

4-39. FLOOR CHANNELS.

4-40. Insertions may be made to floor channelsby cutting away the damaged portion and inserting the necessary length of channel. The channel inserted should be identical to the original and may be spliced to the original as shown in figure B-20.

4-41. HAT SECTIONS.

4-42. Damaged hat sections which are not repairable by patching may be repaired by trimming away the damaged areas and inserting a length of hat section. Splices at each end of the insertion may be made as shown in figure B-16.

4-43. SKIN.

4-44. Skin damage which is too large to be repaired by patching must be repaired by trimming the skin back to structural members and inserting a section of skin. The skin in the adjacent panels will be allowed a minimum of 5/8-inch edge distance beyond the centerline of the rivets bounding the respective panels. Replacement skin must be of the same material as original, or stronger, and must be of a thickness equal to or a gauge heavier than the original. Rivet spacing should be accomplished as shown in figure B-8.

4-45. SUPPORTING STRUCTURES.

4-46. In the event of damage to supporting structure which is too large to be restored by a patch, form a replacement insertion of material equal to the original and rivet into place.

4-47. DOOR AND WINDOW FRAMES.

4-48. The cabin window framework is made of hat sections and extrusions, insertions may be made in the same manner as described for hat sections and extrusion repairs. The cabin and baggage door frame construction is similar to that of the bulkheads and they may be repaired in the same manner.

4-49. DAMAGE REPAIRABLE BY REPLACEMENT.

4-50. BULKHEADS.

4-51. Damage to bulkheads must be very carefully considered. Many of the bulkheads are built in sections and replacement of damaged sections is relatively easy; however, some of the bulkheads are difficult to replace, involving the removal of a great deal of equipment. In general, ease of replacement will determine to a large extent whether to repair or replace damaged members.

4-52. STRINGERS.

4-53. Unless the complete stringer is damaged, there is no need to replace the stringer, as patch and insertion repairs may be made easily and are equally satisfactory.

4-54. FLOOR CHANNELS.

4-55. In event of extensive damage to the floor channels, it will be more satisfactory to replace them than to attempt a repair. The cross member of Bulkhead 4, Beech Part 804-184047, (see figure 4-3), is a very highly stressed member constructed of chrome-molybdenum steel, either .063 sheet or .065 strip, Specification AN-QQ-S-685. Replacement of this member is recommended if it is damaged.

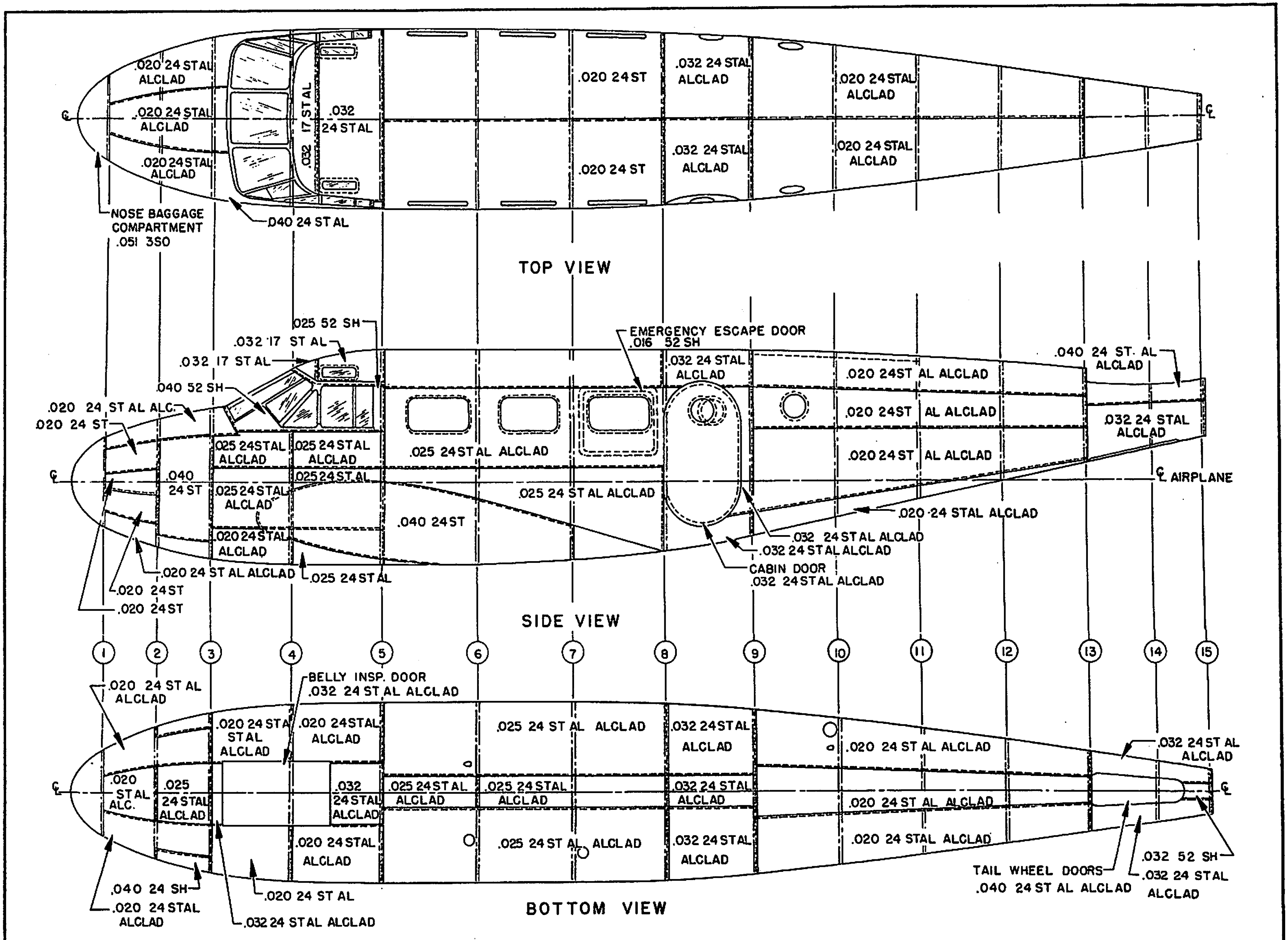


Figure 4-2. Fuselage-Skin Plating Diagram

4-56. HAT SECTIONS.

4-57. Replacement of hat sections will be determined by the amount of work necessary to replace them. If a complete hat section can be replaced with a slight amount of additional work, it will be better to replace the damaged part.

4-58. SKIN.

4-59. In sections of skin in which the area of damage is relatively large, the section of skin affected should be replaced. Before removing the skin, determine whether or not the rivets can be bucked on the replacement skin. Where rivets cannot be bucked, in many cases it will be better to make an insertion repair.

4-60. SUPPORTING STRUCTURE.

4-61. Replacement of supporting structure will be determined largely upon the extent of the damage and availability of replacement parts. In general, if replacement parts are available, it will be better to make a replacement than to attempt a repair, even though the damaged area be small. A replacement will not add weight to the aircraft and in most instances will result in a better appearance than that made by other types of repair.

4-62. WINDSHIELD AND WINDOWS.

4-63. PROCEDURE FOR REMOVAL.

a. Remove screws holding lower gusset, outside "T" frame and upper strip.

NOTE

Access to the elastic stop nuts is obtained through the nose baggage compartment and the pilot's compartment.

b. Using a sharp knife to loosen sealer carefully remove lower gusset, outside "T" frame and upper strip.

c. Remove glass and clean all sealer from channels and surfaces.

4-64. INSTALLATION.

4-65. PROCEDURE FOR INSTALLATION.

a. Install sealer, using a double thickness of 1/2-inch wide presstite No. 155, along lower surface of inside "T" frame and upper and lower channel.

b. Install glass in place leaving 1/8-inch clearance between edge of glass and channels; and press down firmly.

c. Install sealer, using a double thickness of 1/2-inch wide presstite No. 155, on the outboard side of the glass near the edge on all edges.

d. Install outside "T" frame, lower gusset and upper strip with screws and tighten securely but do not overtighten. Trim off excess sealer and clean glass.

NOTE

Screws in the glass frame are to contact the fiber in the stop nuts at least two threads and are not to protrude through fiber more than one thread.

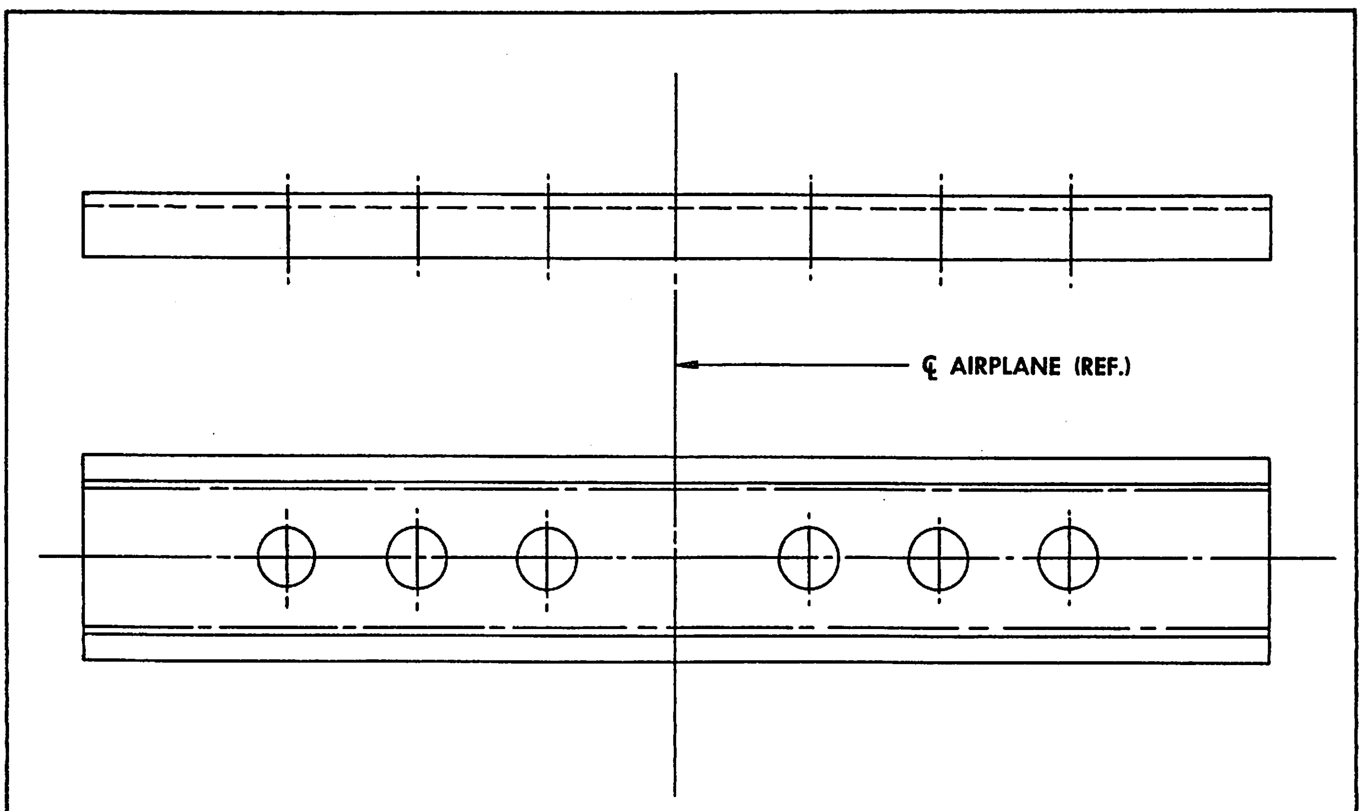


Figure 4-3. Cross Member, Bulkhead 4

SECTION V

ALIGHTING GEAR

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5-1. MAIN LANDING GEAR.

5-2. DESCRIPTION.

5-3. The main landing gear is an air-oleo shock strut, having a yoke-type lower end clamped rigidly to the axle of the wheel. At the upper and lower ends of the cylinder, the strut is attached to a V-brace pivoting on bolts through bronze bushings mounted in the center section steel truss (see figure 5-1). Hinged to the rear side of the lower end of the upper cylinder is an oleo drag strut running aft and upward. The upper end of the drag strut is hinged to a slide operating on a slide tube which runs forward and upward in approximately the center of the engine nacelle, from the bottom of the center section truss to a point near the top of the firewall. Attached to the slide is a chain which, when operated by the landing-gear motor or emergency hand crank through the retraction mechanism, pulls the slide forward and up the slide tube, retracting the landing gear aft and up into the bottom of the nacelle.

5-4. TAIL GEAR.

5-5. DESCRIPTION.

5-6. The tail gear consists of a fork, swivel housing, air-oleo shock absorbing unit, wheel, tire, and locking mechanism (see figure 5-2) essentially as follows:

a. The fork, which holds the wheel assembly, swivels in the swivel housing which is hinged for retraction of the front end in fittings provided on a fuselage bulkhead.

b. The rear end of the swivel housing is bolted to the lower end of the shock strut.

c. The upper end of the shock strut is bolted to a slide operating on a fore-and-aft tube. When the landing gear is retracted, the slide is pulled forward on the slide tube, retracting the tail gear aft and up into the fuselage.

d. The fork is of welded steel tubing and gusset construction. The tail wheel mounts in the fork.

e. The swivel housing also is of welded steel tubing and gusset construction. It houses the swivel end of the fork and is hinged at the front end on an aluminum alloy shear bolt through fittings attached to a fuselage bulkhead.

5-7. DEFINITION OF DAMAGE.

5-8. Damage may be defined as follows:

a. Negligible damage is damage that will not affect the airworthiness of the main landing gear or the tail wheel gear, and does not require immediate attention.

b. Damage repairable by patching is damage that may be repaired by covering or reinforcing without removal of a portion of the main landing gear or tail wheel gear structure.

c. Damage repairable by insertion is damage requiring the removal and replacement of a portion of the main landing gear or tail wheel gear structure.

d. Damage repairable by replacement is damage unrepairable by patching or insertion, but that may be repaired by installing a new part.

5-9. PROCEDURE FOR REPAIR OF DAMAGE.

5-10. NEGLIGIBLE DAMAGE.

5-11. MAIN LANDING GEAR.

5-12. Negligible damage may consist of one, or both, of the following conditions:
a. Scratches, not exceeding 1/32-inch in depth, and not more than five of them occurring in one square inch of surface.
b. Dents, not exceeding 1/32-inch in depth, having no more than 3/32-inch radii, and no more than four of them occurring in one square inch of surface. THIS METAL SURFACE MUST NOT BE CRACKED OR FRACTURED.

NOTE

Damage of the above nature does not require immediate attention; however, it is recommended that scratches be kept free of sharp, rough edges, and that they be pointed.

5-13. TAIL GEAR.

5-14. Negligible damage to this structure may take the form of slight indentations, scratches, or minor bowing. Smooth dents not exceeding one-twentieth of the tube diameter in depth without cracks, fractures, or sharp corners, and clear of the middle third of the length of the member may be disregarded except to satisfy appearance. Tubular members should be carefully examined for the presence of sharp nicks and deep scratches, because these nicks and scratches produce stress concentrations that may cause failure of the part.

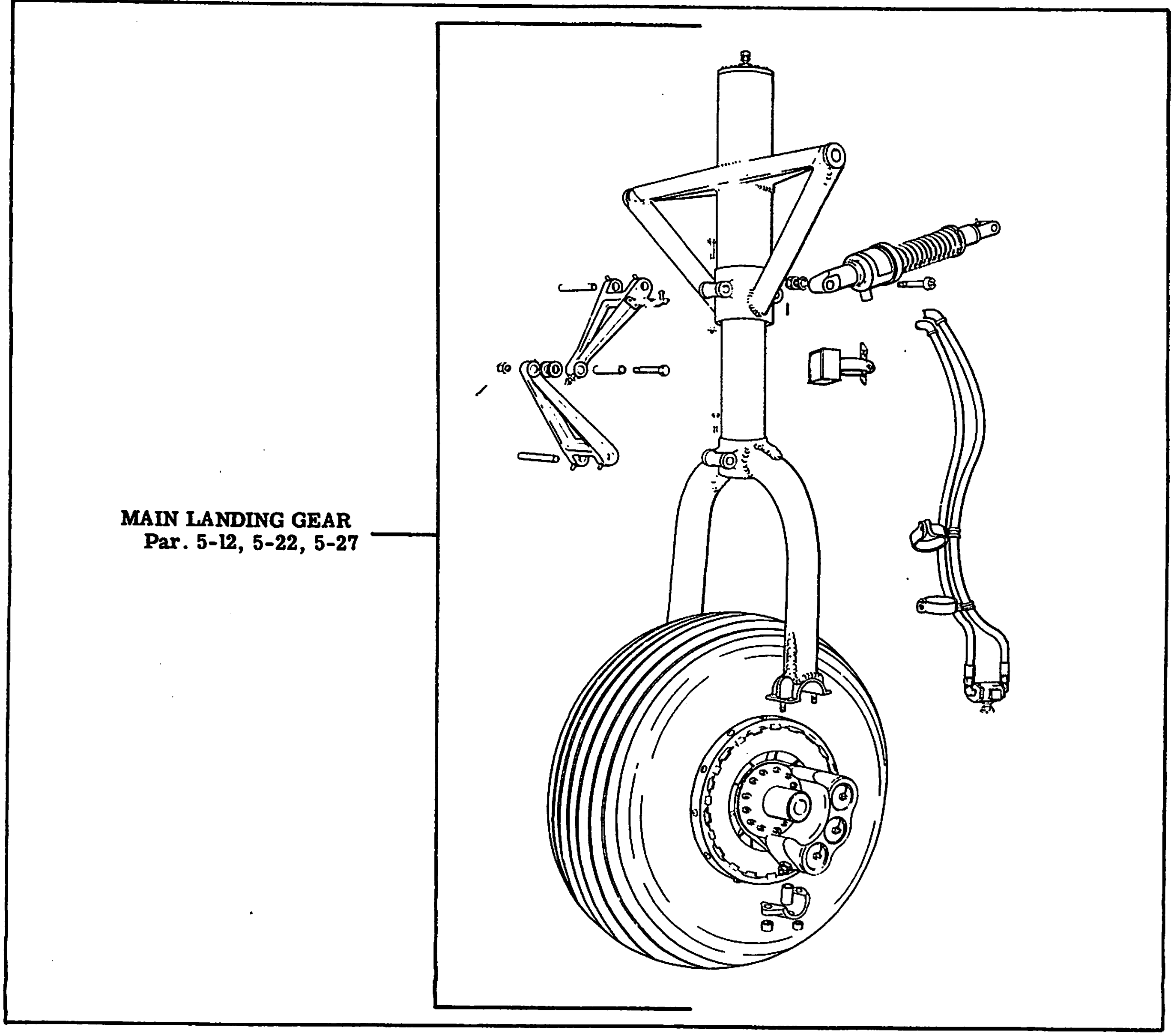


Figure 5-1. Main Landing Gear Exploded

5-15. DAMAGE REPAIRABLE BY PATCHING.

5-16. MAIN LANDING GEAR.

5-17. The structural members comprising the main landing gear are heat-treated and highly stressed; therefore, repairs by welding are prohibited.

5-18. TAIL GEAR.

5-19. The tail wheel swivel housing assembly may be repaired by patching. Since the tail wheel swivel housing assembly is a normalized structure, this is recommended as a temporary repair only, and the assembly should be replaced as soon as possible.

5-20. DAMAGE REPAIRABLE BY INSERTION.

5-21. MAIN LANDING GEAR.

5-22. Repair of any damage in excess of that listed as negligible damage, paragraph 5-12, is prohibited.

5-23. TAIL GEAR.

5-24. The tail wheel swivel housing assembly may be repaired by splicing or partial replacement of the tubular structure. In making the repair, be sure the function of moving parts is not impaired.

5-25. DAMAGE REPAIRABLE BY REPLACEMENT.

5-26. MAIN LANDING GEAR.

5-27. Any damage sustained in excess of that listed as damage repairable by patching in paragraph 5-18 will necessitate replacement of the damaged part.

5-28. TAIL GEAR.

5-29. Any damage sustained by the tail wheel fork assembly, other than that described in paragraphs 5-14, 5-19, and 5-24, will necessitate replacement of the fork assembly. This assembly consists of the fork to which the tail wheel is mounted, and the barrel, or swivel, which is the upper part of the fork and which turns in the swivel housing. This fork assembly is highly stressed and constructed of heat-treated steel tubing. It cannot be repaired because any bending or welding would greatly reduce its strength.

NOTE

The tail gear retract cable is attached to the slide assembly by a shear bolt. If repairs are accomplished which involve disconnecting the cable and slide, make certain the same bolt is used to reconnect them. Under no circumstances should a steel bolt be used for this purpose.

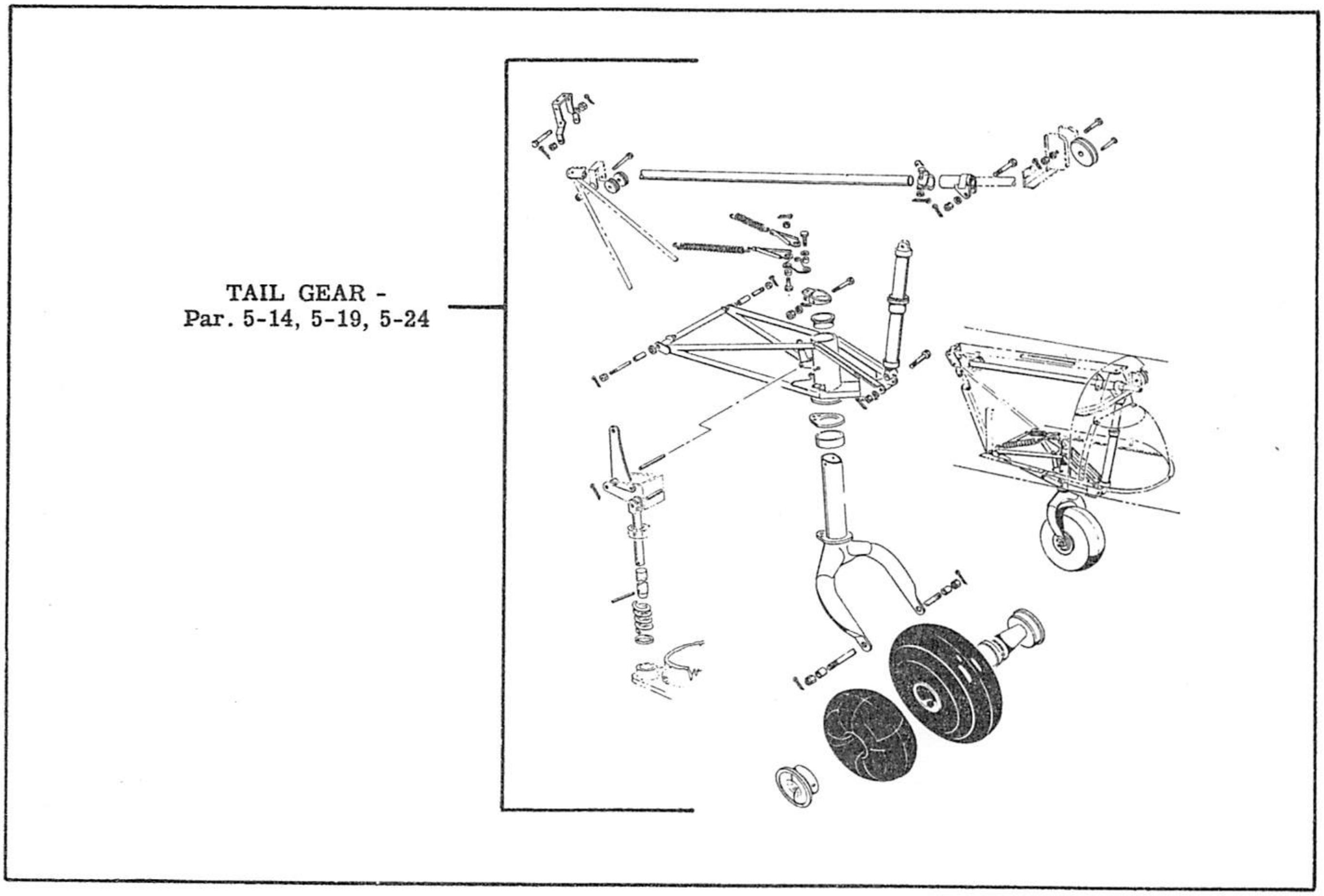


Figure 5-2. Tail Gear, Exploded

SECTION VI

NACELLE GROUP

TABLE OF CONTENTS

6-1. NACELLE GROUP.

6-2. DESCRIPTION.

6-3. The nacelle is built integrally with the center section wing (see figure 6-1). The aft portion of the nacelle, which provides fairing for the engine and housing for the landing gear and oil supply tank, is a skin-covered, bulkhead-supported structure. The lower side is enclosed by hinged landing gear doors.

6-4. COWLINGS.

6-5. DESCRIPTION.

6-6. The accessory compartment of the engine section is separated from the engine by an inner cowl (see figure 6-1), and the entire section is enclosed by the engine ring cowling which is connected by adjustable strap fasteners. The upper section of the cowling extends to the firewall, covering the engine accessory compartment. The lower section covers only the engine and is fitted with flaps for engine temperature control. The cowling is supported by brackets on the engine cylinders and a former ring at the firewall. Dzus-type fasteners hold the sections in assembly.

6-7. WRAPPER SHEETS.

6-8. DESCRIPTION.

6-9. The engine wrapper sheets cover the lower section of the engine compartment. They are in two sections, fastened with Dzus and strap fasteners.

6-10. COWL FLAPS.

6-11. DESCRIPTION.

6-12. Three separate, overlapping cowl flaps are fitted to the trailing edges of the lower section of the ring cowl, on each side of the engine. They are hinged at the forward edge and interconnected so they operate as a unit (see figure 6-1).

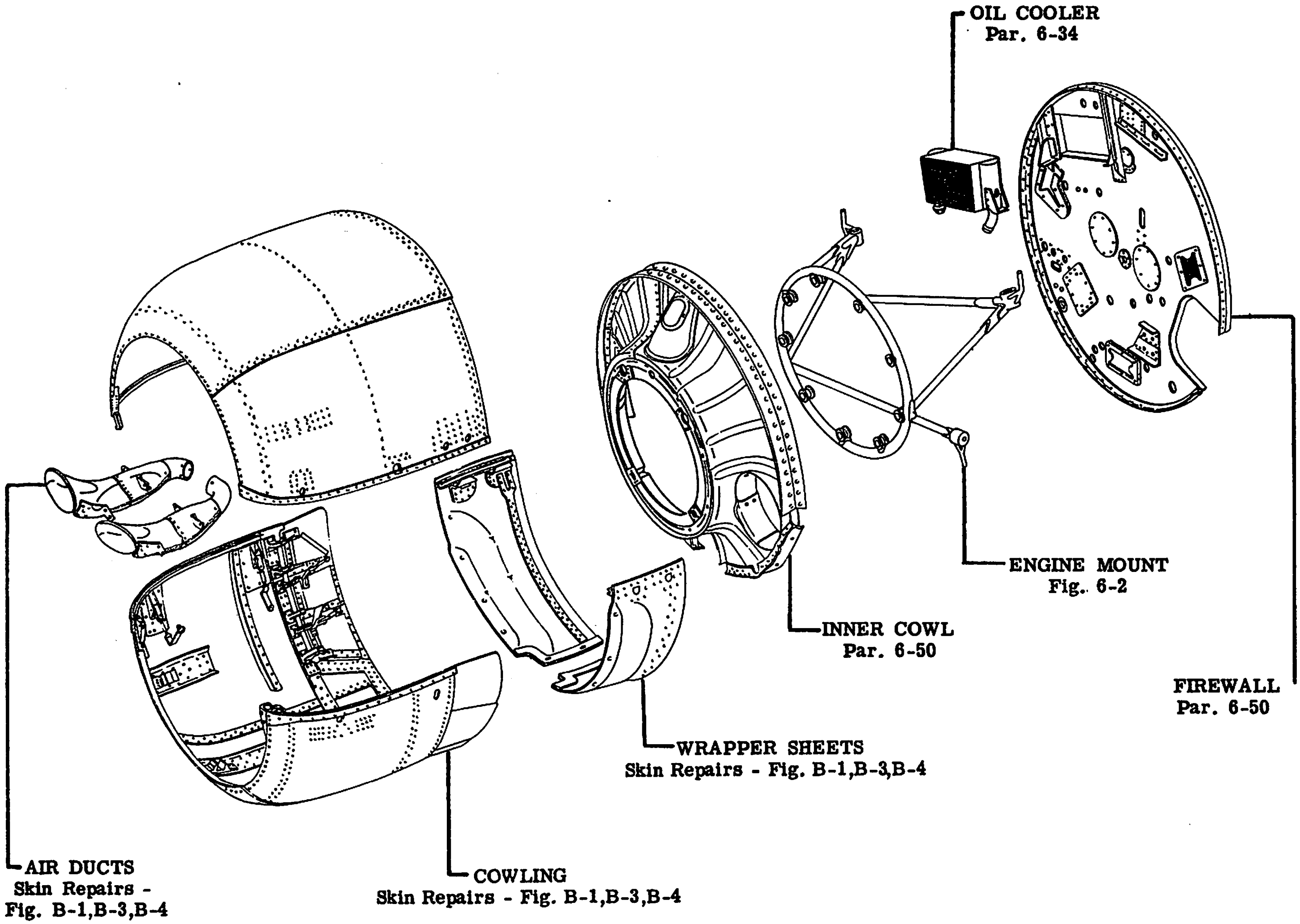


Figure 6-1. Nacelle Group Exploded

6-13. FIREWALL.

6-14. DESCRIPTION.

6-15. The forward bulkhead of the nacelle is a stainless steel firewall, separating the aft nacelle section from the engine section (see figure 6-1).

6-16. ENGINE MOUNTS.

6-17. DESCRIPTION.

6-18. The engine mount is a normalized, welded assembly of chrome molybdenum steel tubing (see figure 6-1). It attaches to the engine with nine bolts which pass through lugs on the engine mount fitting. The engine mount attaches to three mounting lugs which are welded to the center section truss and extend forward through the firewall. Lord-type shock mounts, pressed into each attaching point on the engine mount, are used to absorb vibration.

6-19. AIR DUCTS.

6-20. DESCRIPTION.

6-21. Air intake ducts for the carburetor are located at the front of the engine near the bottom. Air enters the ducts, then passes through screen filters before reaching the carburetor.

6-22. OIL COOLER.

6-23. DESCRIPTION.

6-24. The honeycomb-type oil cooler is attached by two brackets to the front of the firewall. Cooling air is supplied by a duct from the leading edge of each inboard wing. If damage occurs, the cooler must be sent to a designated overhaul activity for repairs.

6-25. DEFINITION OF DAMAGE.

6-26. Damage to the nacelle group may be defined as follows:

a. Negligible damage is damage not affecting the airworthiness of the nacelle group; does not require attention, other than investigation.

b. Damage repairable by patching is damage repairable by covering or reinforcing any of the component parts of the nacelle group.

c. Damage repairable by insertion is damage requiring the removal and replacement of a portion of any component part of the nacelle group.

d. Damage repairable by replacement is damage requiring the replacement of component parts or assemblies.

6-27. PROCEDURE FOR REPAIR OF DAMAGE.

6-28. NEGLIGIBLE DAMAGE.

6-29. Small holes and dents in the skin, bulkhead structures, landing gear door channels, cowlings, or ducts may be considered negligible if not located so as to weaken a structural part. Dents should be thoroughly investigated for indication of structural damage. Holes must not be over 1/2-inch in diameter, must be rounded and free from ragged edges. It is safer practice to patch or reinforce all holes. Slight indentations, scratches, and smooth dents in the engine mount not exceeding one-twentieth of the tube diameter in depth, and without cracks, fractures or sharp corners, may be considered negligible. Tubular members should be carefully examined for the presence of sharp nicks and deep scratches which produce stress concentrations that may cause failure of the part. Care must be taken to smooth out all sharp nicks and deep scratches with a fine file. When this is done, high concentrations of stress disappear.

6-30. DAMAGE REPAIRABLE BY PATCHING.

6-31. SKIN.

6-32. Small holes, breaks, or cracks in the skin covering and landing gear doors may be repaired by the following methods:

a. Round holes up to one-inch in diameter may be patched by covering with a disc, as shown in figure B-4. If the rivets are not accessible for bucking, a washer may be substituted for the disc, or Cherry blind rivets used with the disc patch.

b. Small breaks or punctures may be patched as shown in figures B-2 and B-3. The ragged edges should be trimmed, as shown by the dotted lines, so that no sharp corners remain.

c. Small cracks should have stop holes drilled at each end or sharp corner before patching (see figure B-1). The patch plate should be cut large enough to clear all dimensions of the crack by at least 3/4-inch. Cherry rivets may be substituted for standard rivets only in areas where it is impossible to buck standard rivets.

6-33. BULKHEADS.

6-34. Small holes, breaks, or cracks may be repaired by methods described for skin patches in the preceding paragraph. Damage in the formed flange (or in the flange and extending down into the web) will require a patch formed to the flange of the bulkhead, similar to the rib repair shown in figure B-14. The crack or break must be covered with a patch of at least 3/4-inch edge margin. Stop holes should be drilled at the ends of cracks, and breaks should be trimmed carefully to prevent spreading.

6-35. FIREWALL.

6-36. Repairs to the firewall will be made similarly to the skin patches in paragraph 6-32. Material for the patches should be of corrosion-resistant steel. Use AN430-AD4 rivets and wet with zinc chromate primer before driving, to minimize corrosion between the aluminum rivets and the steel firewall.

6-37. LANDING-GEAR DOOR CHANNELS.

6-38. Small holes, cracks, and breaks in door channels may be repaired by forming a splice-plate patch of similar material and cross section as the original channel. Use AN430-AD4 rivets with 3/8-inch spacing.

6-39. ENGINE MOUNT.

6-40. If a crack appears in a steel tube (usually as the result of previously straightening the tube), first drill a No. 40 (.098) hole at each end of the crack, then with steel wool remove the finish around the tube for about three inches on each side of the damage. If the damage is in the form of a sharp dent which cannot be removed by any of the methods previously outlined, remove the finish in the above manner and reinforce the tube. Select a length of steel tube having the same wall thickness and an inside diameter approximately the same as the outside diameter of the damaged tube. Cut the sleeve on a 30-degree angle at both ends so it extends beyond the crack or dent not less than 1-1/2 times the diameter of the tube (see figure 6-2). Cut through the entire length of the sleeve and separate the halves. Clamp the two halves in position, over the damaged area. Weld the sleeve along the two sides, then weld both ends to the damaged tube. Torch normalize after welding.

CAUTION

Make no splice by butt welding member between station points.

6-41. COWLING.

6-42. Holes and cracks in the cowling assemblies may be repaired by patching procedures similar to those used on the wings and fuselage, except that fewer rivets are required due to reduced loads. (All holes should be trimmed and cracks stop drilled to prevent spreading.) Patches to the outer sections of the cowling should be of the flush type as shown in figure B-3.

6-43. AIR DUCTS.

6-44. Holes and cracks in the duct assemblies may be patched by any method to cover the damage. Holes should be trimmed and cracks stop drilled to prevent spreading, since the ducts are subject to intense vibration from the engine.

6-45. DAMAGE REPAIRABLE BY INSERTION.

6-46. SKIN.

6-47. Insertion repairs to the nacelle skin are not practical. The panels are of such size that replacement would be preferred if the damage was too extensive to be repaired by patching.

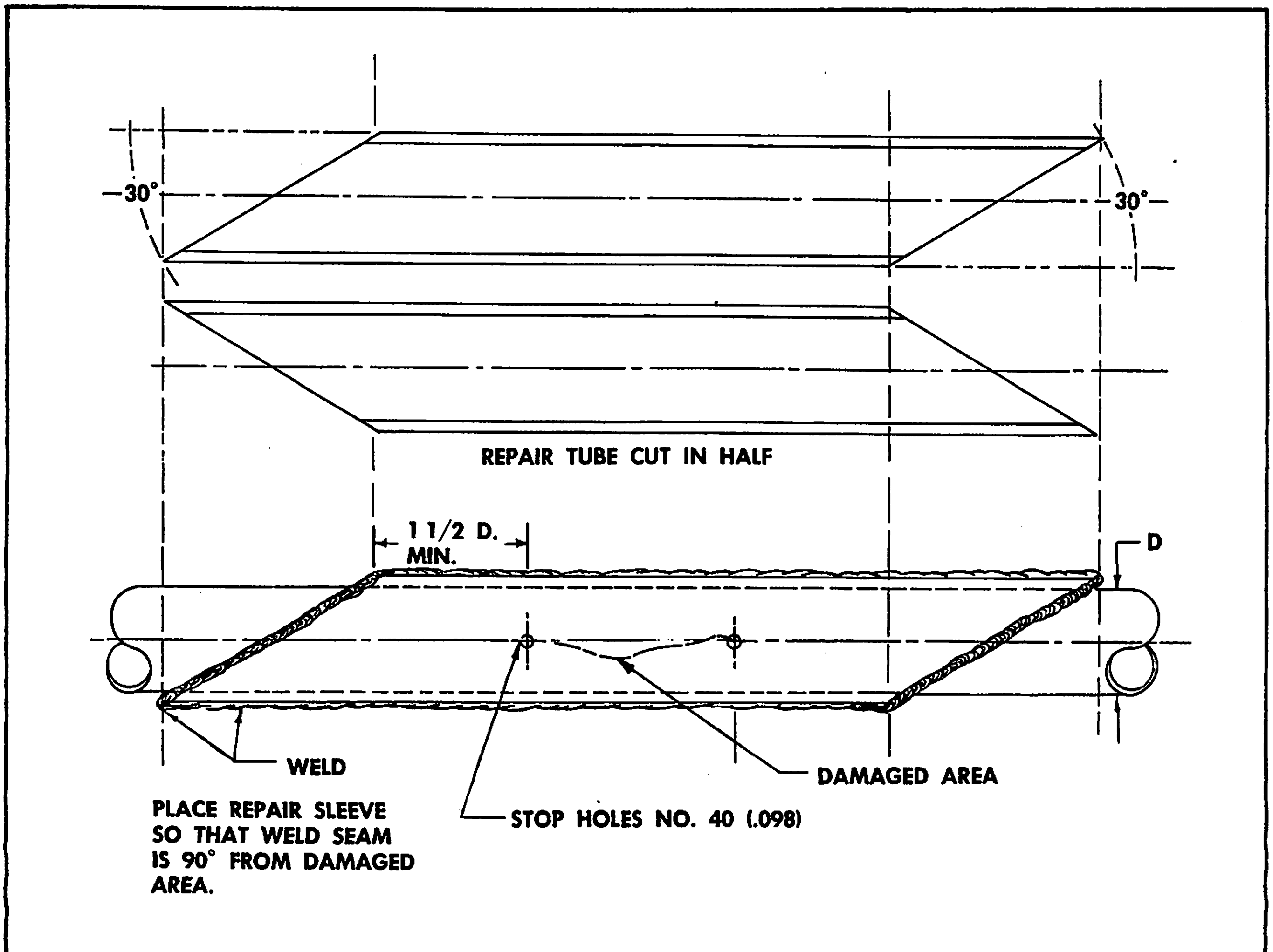


Figure 6-2. Engine Mount Tube Repair

6-48. BULKHEADS.

6-49. Insertion repairs may be made to these members by trimming away the damaged portions and splicing in new sections. New sections may be hand formed from SO material of the next thicker gage and of alloy equal to the original, then heat-treated to the ST condition. If the new section is very large or in the curved portions of the bulkhead, it probably will be easier to obtain the portion needed from a new or salvaged part. The splice plate should be equal to the original material in strength and the next higher gage in thickness. The plate should be at least two inches wide with rivets arranged in two rows and spaced according to procedure in Section I.

6-50. FIREWALL.

6-51. Repairs for the firewall may be made similarly to those for bulkheads. Insertions and splice plates will be of corrosion-resistant steel. Rivets must be dipped in zinc chromate primer before driving.

6-52. LANDING-GEAR DOOR CHANNELS.

6-53. New sections of these channels may be inserted after trimming away the damaged portion. The required portion of channel may be obtained from a new or salvaged part, or short lengths of the channel may be hand formed from SO material and then heat-treated to ST. The splice plate should be of material equal in strength and of the next higher gage in thickness. For rivet arrangement of channel splices, see figure B-18.

6-54. ENGINE MOUNT.

6-55. Insertions may be made to the engine mount structure by welding methods. Splices should not be made in the ring tubing or within three inches of a welded joint.

6-56. COWLING.

6-57. Large holes or dents in the cowling may be repaired by removing the damaged material and inserting new skin. Use SO material of strength and gage equal to the original. Joints, originally spot welded, may be made by riveting.

6-58. DAMAGE REPAIRABLE BY REPLACEMENT.

6-59. SKIN.

6-60. Extensive damage to a skin panel is best repaired by replacement. Select material of original strength and the next gage heavier. Cut and shape as necessary. Follow original rivet pattern in attaching skin panels.

6-61. BULKHEADS.

6-62. If the nacelle bulkheads have incurred as much as 50 percent damage, usually it is advisable to replace the bulkhead with a new part. Follow original rivet pattern when installing bulkhead.

6-63. FIREWALL.

6-64. Replacement of the firewall will be unusual and dependent on the nature of damage and facilities for repair. If it is to be replaced, it will be necessary to remove engine, engine mount, and all connections. Remove rivets attaching front flange to firewall to nacelle skin and Curtis-type clamps holding firewall to truss structure. Follow original rivet pattern when installing new part.

6-65. LANDING-GEAR DOOR CHANNELS.

6-66. Extensive damage to these channels will necessitate replacement in order to maintain proper alignment and operation of the landing-gear doors.

6-67. ENGINE MOUNT.

6-68. Extensive damage to the ring member of the engine mount, damage too close to a welded joint to permit splice repairs, or misalignment of the structure, will necessitate replacement of the entire mount assembly.

6-69. COWLING.

6-70. Damage to any section of the cowling assembly, so extensive that original contour cannot be restored or security of attachment is endangered, will necessitate replacement of that section.

SECTION VII

FABRIC REPAIR AND ATTACHMENT

TABLE OF CONTENTS

7-1. COVERING.

7-2. All flight control surfaces, except trim tabs are fabric covered. The envelope method of covering is used and the attachment procedure used is the same on all control surfaces.

7-3. DEFINITION OF DAMAGE.

7-4. Damage may be defined as follows:
a. Negligible Damage - Damage that will not affect the airworthiness of the fabric and does not require immediate repair.
b. Damage Repairable by Patching - Damage that may be repaired by covering or reinforcing, without removal, other than trimming off a portion of the fabric.
c. Damage Repairable by Insertion - Damage necessitating removal and replacement of a section of fabric.
d. Damage Repairable by Replacement - Damage necessitating complete recovering of the control surface.

7-5. PROCEDURE FOR REPAIR OF DAMAGE.

7-6. NEGLIGIBLE DAMAGE.

7-7. Any damage sustained by a fabric covering should be repaired, however slight it may seem. Damage such as missing drain grommets and loose fabricoid patches, checks, cracks in doped surfaces are not in themselves serious and could be considered negligible damage. Continued use of the aircraft without repair would tend to aggravate the damage and a more serious condition might develop. It is recommended that all fabric damage, however slight it may seem, be repaired at the earliest possible time.

7-8. DAMAGE REPAIRABLE BY PATCHING.

7-9. Damage repairable by patching, such as small holes and tears in the fabric, may be corrected by the accepted fabric-repair methods.

7-10. DAMAGE REPAIRABLE BY INSERTION.

7-11. If an entire fabric section between ribs is damaged to such an extent that the patch will exceed one-half the surface area or that repair patches will overlap, the fabric between ribs should then be replaced.

7-12. DAMAGE REPAIRABLE BY REPLACEMENT.

7-13. In the event a fabric covering is damaged to such an extent that repairs by patching would be impractical, or in the event the fabric loses its tautness and elasticity due to age or numerous repairs, it should be removed and the frame recovered. The envelope method of covering is accomplished by sewing together fabric conforming to Specification AN-C-121, cut in lengths sufficient to pass completely around the frame, starting at and returning to the trailing edge.

NOTE

When recovering elevators, one layer of fabric may be used on both the upper and lower surfaces.

7-14. All sharp metal edges such as lightening holes and flanges which are likely to come in contact with rib lacing must be covered with one-fourth inch cellulose tape at such contacting points.

7-15. Fabric covering on the airfoil is applied so that the warp threads, (threads running parallel to the selvage edges) are parallel to the line of flight.

7-16. If the covering of a component cannot be accomplished without pinning the fabric in place preparatory to hand sewing, one or two wraps of friction tape around the metal part of one- or two- foot intervals will give a surface for pinning to hold the fabric in position for hand sewing.

7-17. Design and sew all coverings so that there will be proper and equal tension over all parts of the sur-

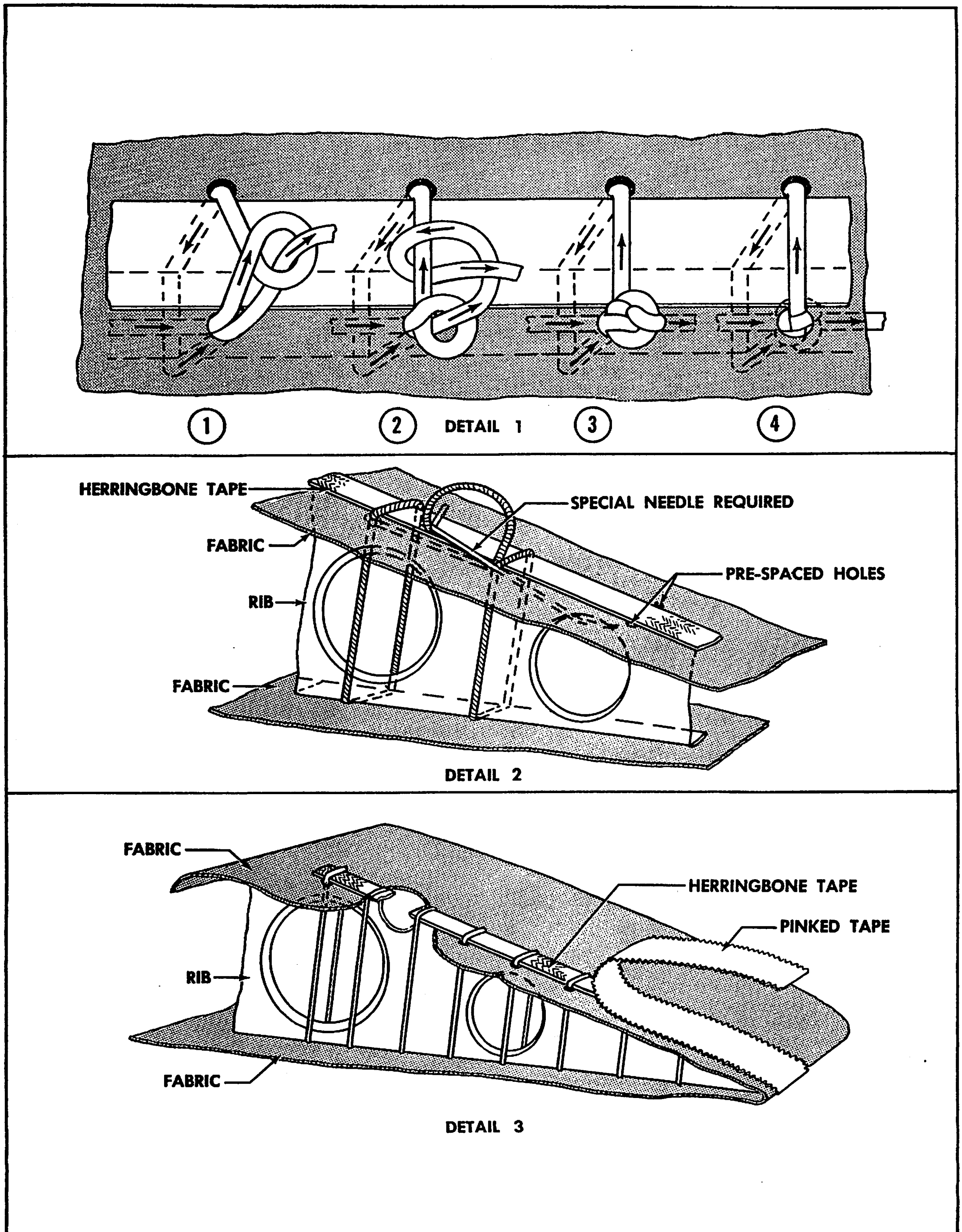


Figure 7-1. Fabric Sewing Procedure

face when the envelope is drawn in position for doping. In order to divide tension among each system of threads in the fabric, equal tension must be applied in all directions.

7-18. Hand sew at a point where machine sewing or uncut fabric is reached. Lock hand sewing at intervals of six inches and finish seams with a lock stitch. At the point where hand sewing is necessary cut the fabric so that it can be doubled under before sewing. In hand sewing maintain a minimum of four stitches per inch. See figure 7-4.

7-19. SEAMS.

7-20. On all seams, use plain lap or folded flat fell seams machine stitched in accordance with Type LSC-2, and a three loop stitch or double locked stitch Type 301 and 401 in accordance with Specification DDD-S-751. On stitch seams use eight to ten stitches per inch. Sew the row of stitches nearest each folded edge of each seam one-eighth to one-quarter inch from the edge of the fold and the rows of stitches one-fourth to three-eighths inch apart.

7-21. Place all longitudinal fore and aft seams parallel to the line of flight. Make certain that seams will not cover a rib or be so placed that the lacing will be through or over a seam. In the case of a tapered section place the seams so as to cross the fewest number of ribs consistent with efficient cutting of the pattern.

7-22. The only permissible seam extending the span of the control surface (laterally) in the envelope, either hand sewed or machine sewed, is the seam at the trailing edge, except in the case of a tapered section, where an additional seam is allowed on the tapered portion at the leading edge. In all cases, cover this seam with a strip of 3-3/4 inch surface tape, USA Specification 6-62.

7-23. REINFORCEMENTS.

7-24. In cases where the fabric is attached by lacing apply reinforcing tape, Specification AN-DDD-T-91, under all lacings (see figure 7-1). Securely attach tape under moderate tension at the forward and rear ends of the fabric-covered portion. Where the attachment of the fabric is made by self-tapping screws, bolts, or other mechanical devices, other types of reinforcements may be used as shown in figure 7-2.

7-25. FABRIC ATTACHMENT.

7-26. Attach the fabric using the flush type attachment as shown in figure 7-2. Use lacing cord and reinforcing tape or mechanical devices as described in paragraph 7-22 provided they are countersunk in grooves of the supporting structure or ribs.

7-27. LACING.

7-28. When lacing thin sections pass the lace completely around the rib cap strip, fabric, and reinforcing tape

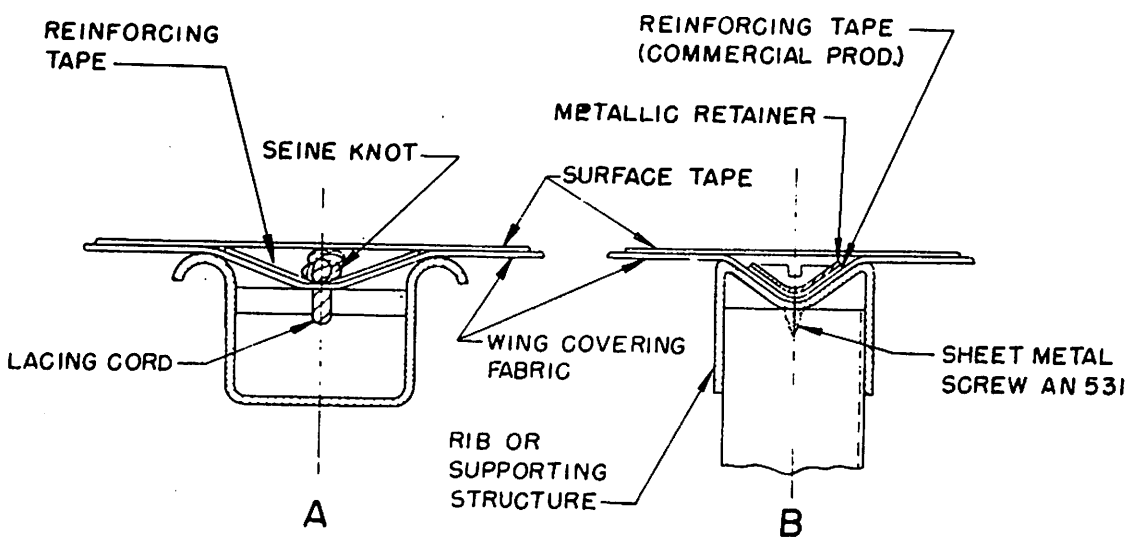


Figure 7-2. Flush-Type Fabric Attachment

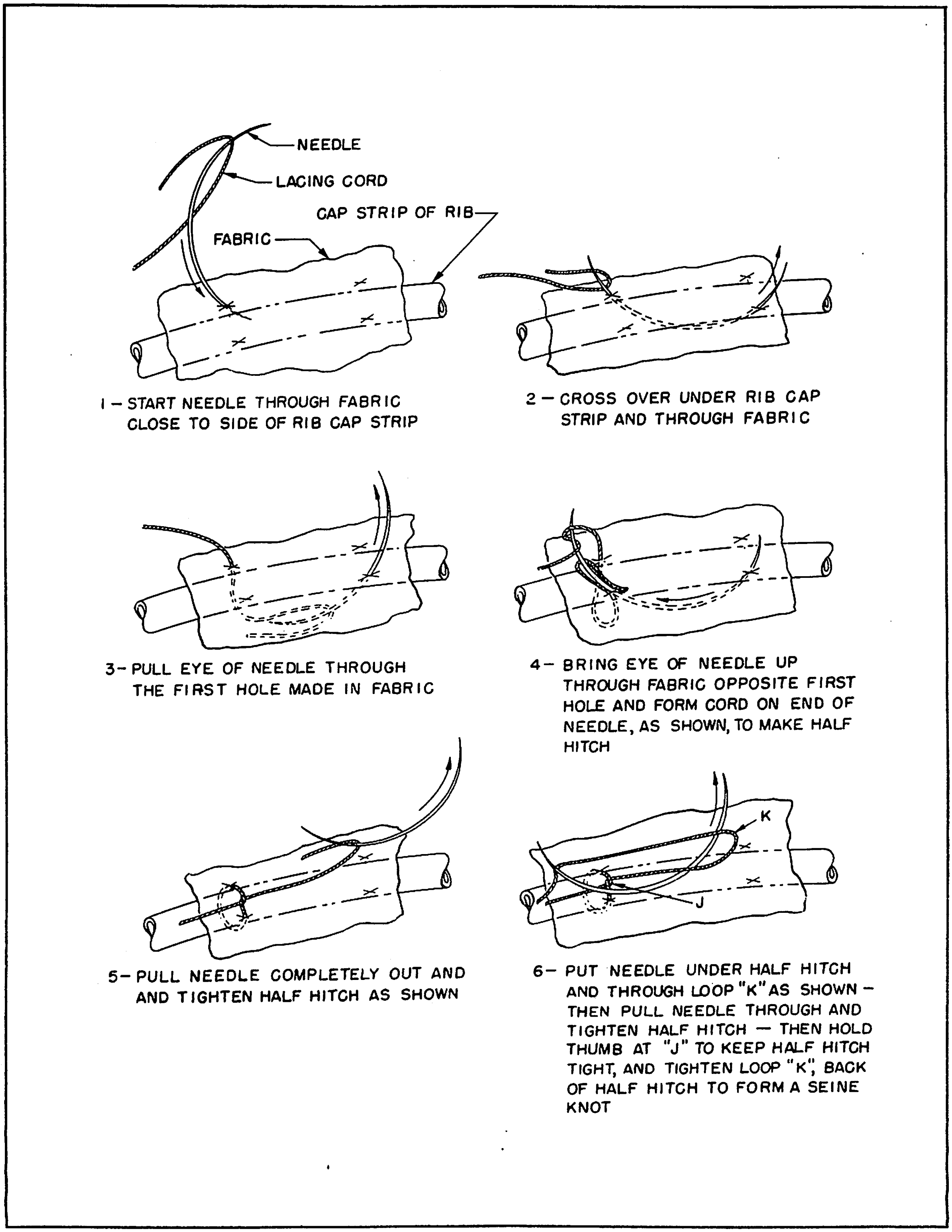


Figure 7-3. Rib Lacing

as shown in figure 7-3. Tie all knots on the upper surface of the airfoil. Apply enough tension on lacing to remove all slack.

7-29. When lacing thick sections, and the rib is designed for lacing, pass the lacing around each upper and lower rib member, fabric, and reinforcing tape as shown in figure 7-1, and 7-3. Lace as close as practicable to the rib upper and lower member and tie over the center or edge of the member.

7-30. Tie a slip knot at the first point of lacing, then carry the cord to the next point or lacing at which, and at all subsequent points, secure the cord at the finish of the lacing by a double or lock knot.

7-31. FINISHING TAPE.

7-32. After the first coat of dope has dried apply the finishing tape, then apply a second coat of dope freely over lacings, reinforcing tape, and that portion of the fabric to be covered by finishing tape, lay the finishing tape over the doped surface and bring into position, then dope freely over the finishing tape.

7-33. Reinforce all portions of fabric that are pierced by wires, bolts or other material after the first coat of dope has dried. To make a reinforcement, dope on an ample patch of fabric. Either pink the patch or fray out on four sides not less than 3/16-inch. Omit this type of patch where other reinforcements are specified. At points where wear or friction is induced by a moving part or fitting, install a cotton duck patch by first sewing the cotton duck to a fabric patch then doping in position.

7-34. Make provision for openings to permit inspection and repair of internal structures and equipment. Use metallic frames for this purpose, attaching the metal inspection doors to the frames.

7-35. Install drainage grommets in each fabric covered component. Place the grommets on the under side at the trailing edge and as close to the rib as practicable. Place a drainage grommet on each side of the rib to insure good drainage. The grommets are made of celluloid and are doped directly to the main covering, after the first coat of dope.

7-36. FINISHING.

7-37. To finish a fabric covered surface proceed as follows:

a. Apply clear dope, Specification AN-TT-D-554. This is applied either by spraying two cross coats or by applying two brush coats then two spray coats.

NOTE

Dope may be thinned as necessary, but every coat must produce a definite film over the previous one.

b. Apply the pigmented dope in such an amount that a solid covering is obtained and the finish has a smooth appearance and feel. Two coats are normally required,

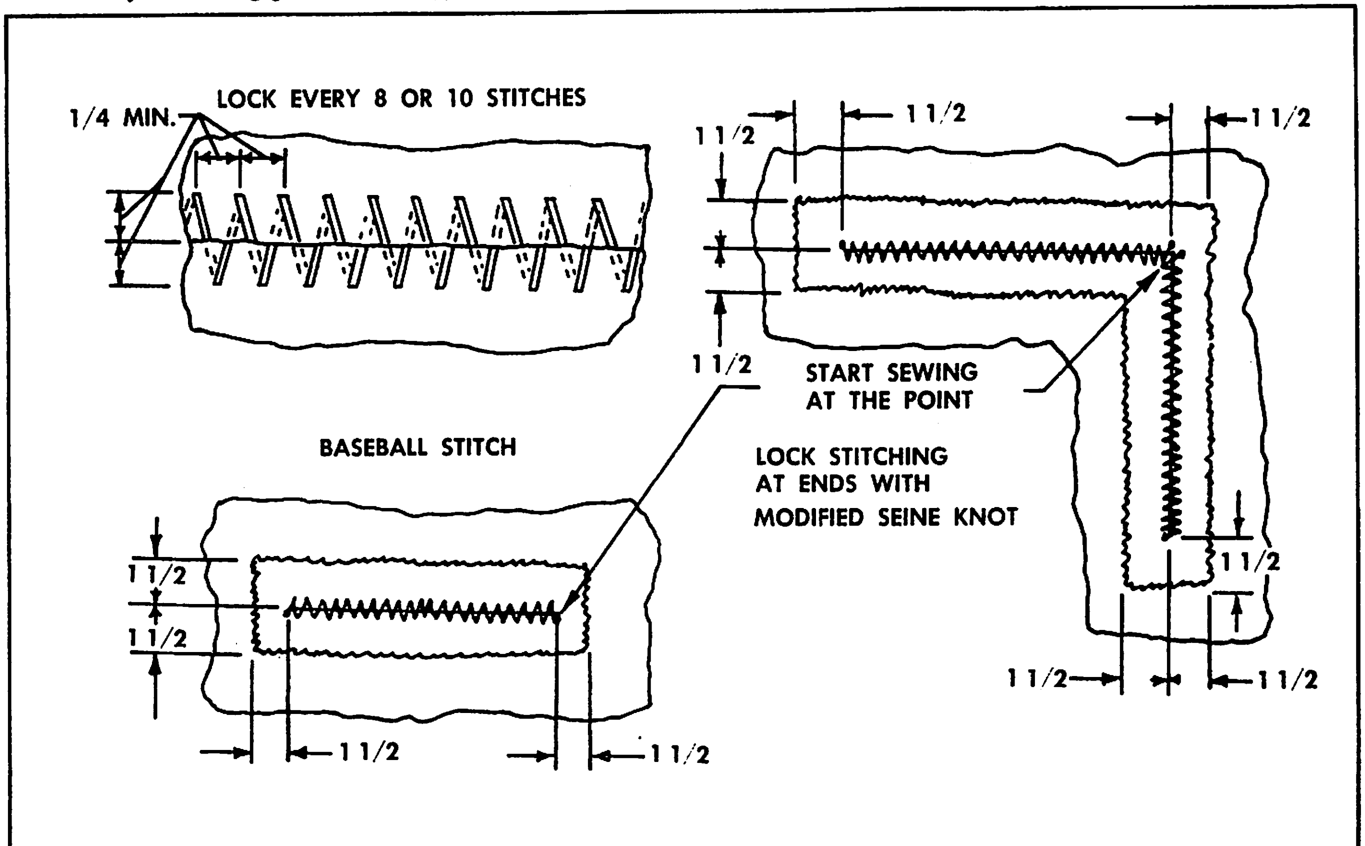


Figure 7-4. Types of Stitches

thoroughly wet spraying the last cross coat.

NOTE

Use thinners Specification AN-TT-T-256 or Specification AN-TT-T-258 as applicable.

7-38. If necessary to produce a smooth finish, sand after the second clear coat and/or after the first pigmented coat using No. 320 grit sandpaper.

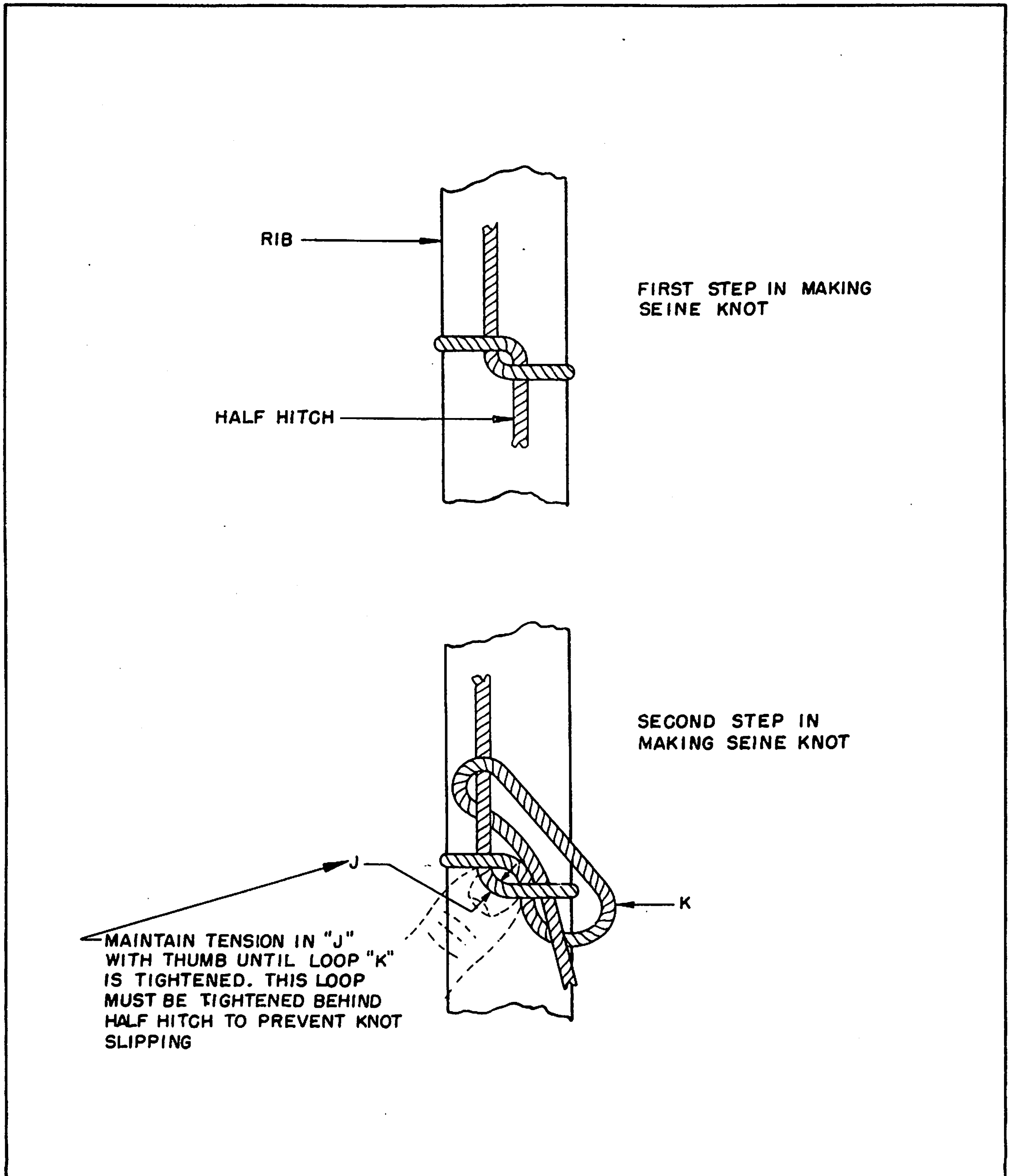
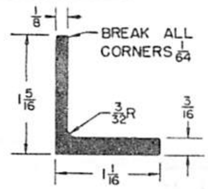
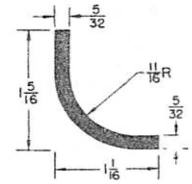
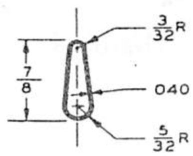
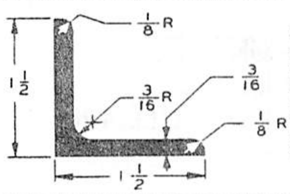
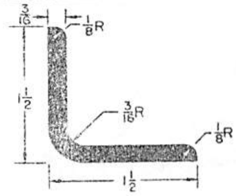
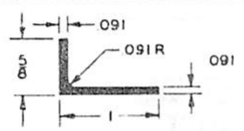
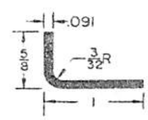
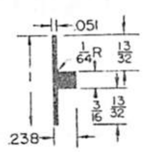
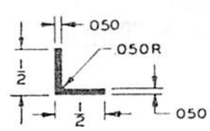
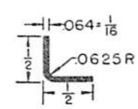
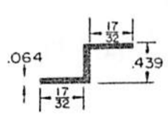
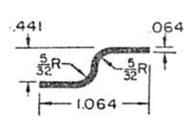
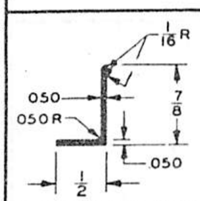
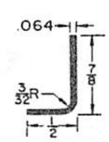
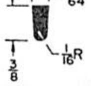
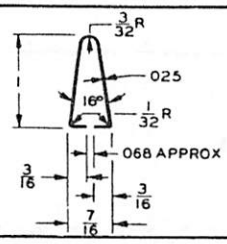
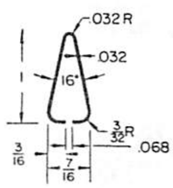
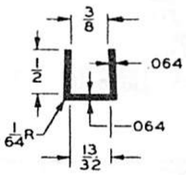
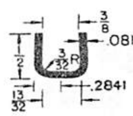
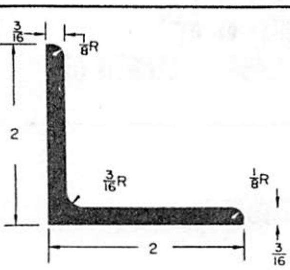
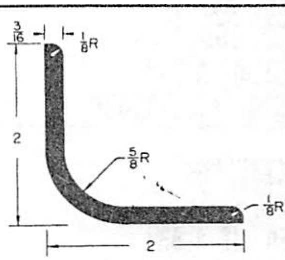
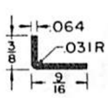
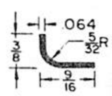
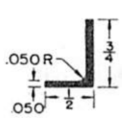
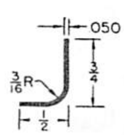
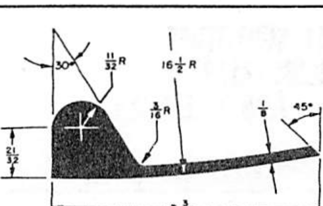


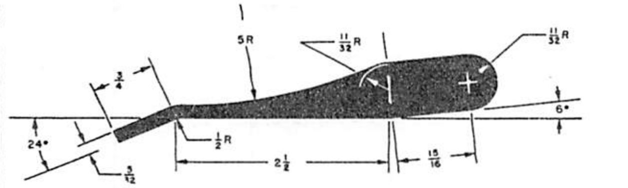
Figure 7-5. Tying a Seine Knot

SECTION VIII

EXTRUSION CHART

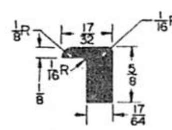
 17S ALUM. ALLOY BOHN ALUM. CO. DIE 86S4 SPEC FED. QQ-A-351 CONDITION T BEECH DWG. NO. 101161	 101161 SUBSTITUTE 24S ALUM. ALLOY 5/32 + 1.877 + LENGTH OF SHEET CONDITION T
 61S TUBING ALCOA DIE T-533 SPEC FED. WW-T-789 BEECH DWG. NO. 101553	101553 NO SUBSTITUTE
 24S ALUM. ALLOY ALCOA DIE 77L SPEC FED. QQ-A-354 BEECH DWG. NO. 101635	 101635 SUBSTITUTE 24S ALUM. ALLOY 3/16 + 2.676 + LENGTH OF SHEET
 24S ALUM. ALLOY ALCOA DIE 15514 SPEC FED. QQ-A-354 BEECH DWG. NO. 101636	 101636 SUBSTITUTE 17SO ALUM. ALLOY .091 + 1.469 + LENGTH OF SHEET
 63ST EXTRUSION ALCOA DIE D17179 SPEC FED. QQ-A-331 BEECH DWG. NO. 101637	101637 NO SUBSTITUTE
 24S ALUM. ALLOY ALCOA DIE K-11623 SPEC FED. QQ-A-354 CONDITION T BEECH DWG. NO. 101639	 101639 SUBSTITUTE 24S ALUM. ALLOY .064 + .89 + LENGTH OF SHEET CONDITION T
 53S ALUM. ALLOY ALCOA DIE D17940 SPEC FED. QQ-A-331 BEECH DWG. NO. 101640	 101640 SUBSTITUTE 61SW ALUM. ALLOY .064 + 1.3655 + LENGTH OF SHEET
 24S ALUM. ALLOY ALCOA DIE K-11622 SPEC FED. QQ-A-354 BEECH DWG. NO. 101641	 101641 SUBSTITUTE 24S ALUM. ALLOY .064 + 1.29 + LENGTH OF SHEET

 63ST EXTRUSION SPEC FED. QQ-A-331 ALCOA DIE K 16822 BEECH DWG. NO. 101644	101644 NO SUBSTITUTE
 24 STAL ALCLAD ALCOA DIE K-1508 CONDITION T BEECH DWG. NO. 101925	 101925 SUBSTITUTE 24S ALUM. ALCLAD .032 + 2.486 + LENGTH OF SHEET CONDITION T
 24S ALUM. ALLOY ALCOA DIE NO. 875 SPEC FED. QQ-A-354 BEECH DWG. NO. 102718	 102718 SUBSTITUTE 17SO ALUM. ALLOY .018 + 1.70 + LENGTH OF SHEET
 17S ALUM. ALLOY ALCOA DIE 77S SPEC FED. QQ-A-351 CONDITION T BEECH DWG. NO. 103287	 103287 SUBSTITUTE 17S ALUM. ALLOY 3/16 + 3.486 + LENGTH OF ANGLE CONDITION T
 17S ALUM. ALLOY ALCOA DIE K-206B SPEC FED. QQ-A-351 BEECH DWG. NO. 104521	 104521 SUBSTITUTE 17S ALUM. ALLOY .064 + .836 + LENGTH OF SHEET CONDITION T
 24S ALUM. ALLOY REYNOLDS METAL CO. DIE E-1213 SPEC FED. QQ-A-351 BEECH DWG. NO. 104588	 104588 SUBSTITUTE 24S ALUM. ALLOY .050 + 1.105 + LENGTH OF SHEET CONDITION T
 24ST ALUM. ALLOY BOHN ALUM. & BRASS CORP. DIE 10013 SPEC FED. QQ-A-354 BEECH DWG. 105410	105410 NO SUBSTITUTE



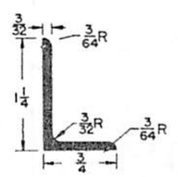
24ST ALUM. ALLOY
ALCOA DIE B10014
SPEC FED. QQ-A-354
BEECH DWG. NO. 105411

105411 NO SUBSTITUTE

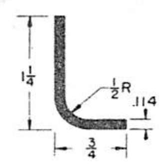


53S ALUM. ALLOY
SPEC FED. QQ-A-331
CONDITION F
BEECH DWG. NO. 106237

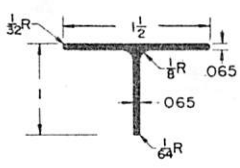
106237 NO SUBSTITUTE



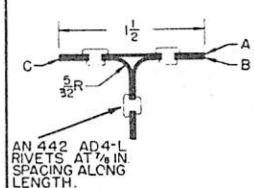
24S ALUM. ALLOY
ALCOA DIE 734FF
SPEC FED. QQ-A-354
BEECH DWG. NO. 110205



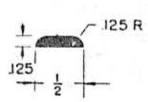
110205 SUBSTITUTE
24S ALUM. ALLOY
.114 + 1.637 + LENGTH OF SHEET
CONDITION T



24ST ALUM. ALLOY
ALCOA DIE 22008
SPEC FED. QQ-A-354
BEECH DWG. NO. 111164

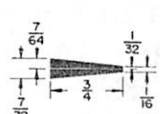


111164 SUBSTITUTE
A — 24ST ALUM. ALLOY
.036 + 1.5 + LENGTH OF SHEET
B AND C —
24ST ALUM. ALLOY
.036 + 1.3755 + LENGTH OF SHEET

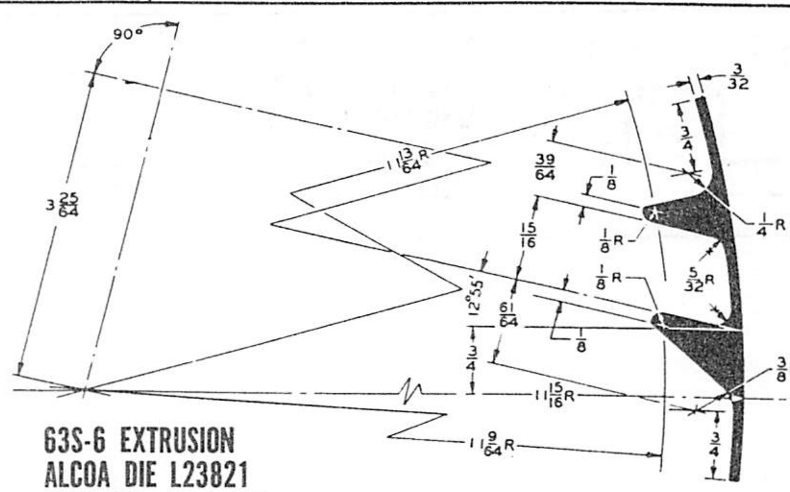


61S ALUM. ALLOY
ALCOA DIE NO. 30319
SPEC FED. QQ-A-325
BEECH DWG. NO. 112139

112139 NO SUBSTITUTE

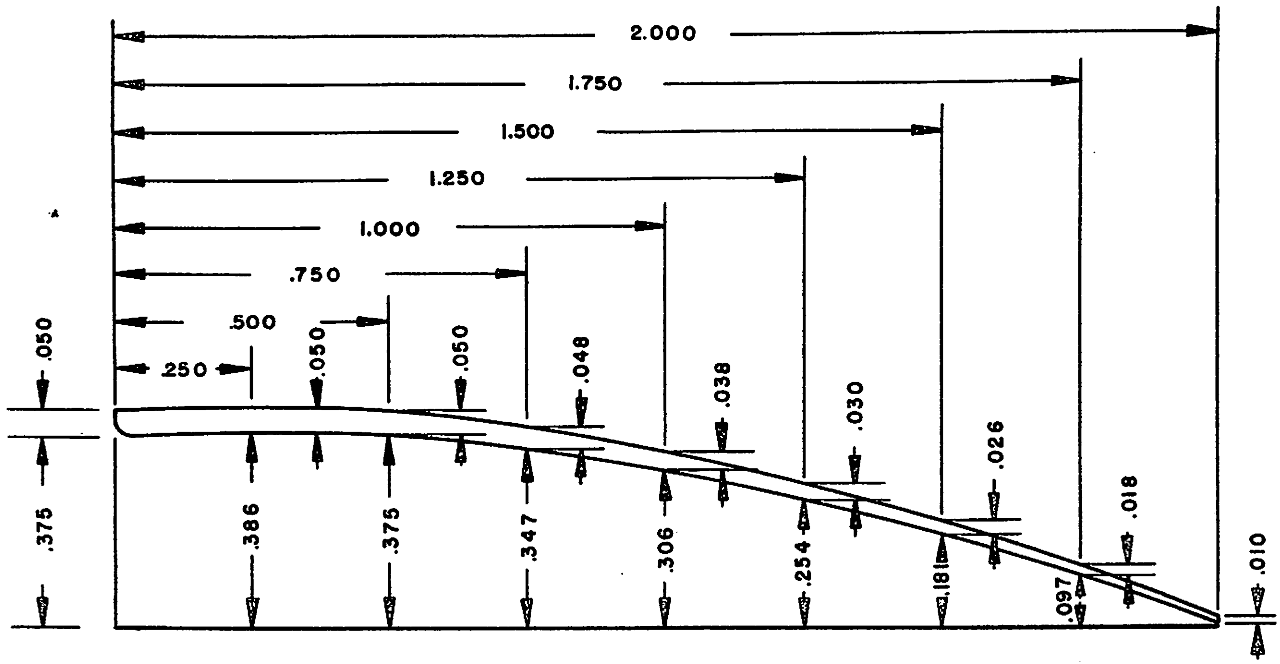


53ST EXTRUSION
ALCOA DIE D20030
SPEC FED. QQ-A-331
BEECH DWG. NO. 186258



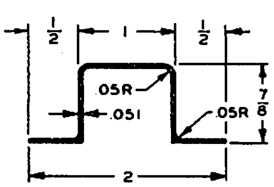
63S-6 EXTRUSION
ALCOA DIE L23821
SPEC FED. QQ-A-331
CONDITION T
BEECH DWG. NO. S-160

S-160 NO SUBSTITUTE

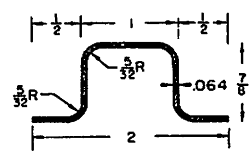


63ST-6 EXTRUSION
ALCOA DIE D-20720
BEECH DWG. NO. S-161

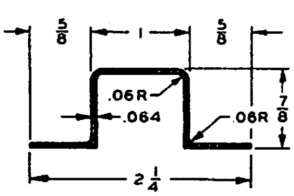
S-161 NO SUBSTITUTE



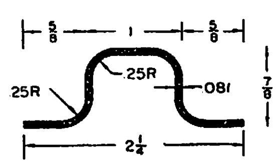
24S ALUM. ALLOY
ALCOA DIE K-22783
SPEC FED. QQ-A-354
CONDITION T
BEECH DWG. NO. S-215



S-215 SUBSTITUTE
24S ALUM. ALLOY
.064 + 3.578 + LENGTH
OF SHEET
CONDITION T



24ST ALUM. ALLOY
ALCOA DIE K-22784
SPEC FED. QQ-A-354
CONDITION T
BEECH DWG. NO. S-216



S-216 SUBSTITUTE
24S ALUM. ALLOY
.081 + 3.972 + LENGTH
OF SHEET
CONDITION T

SECTION IX

TABLE OF HEAT TREATED FITTINGS

Part Number	Name	H.T., Lb/Sq. in.	Com'l Design	Specification	Material
404-184138 & -1	Fitting - Tailwheel truss pivot pin bearing	95,000 to 135,000	MS 191*		Chrome-Moly Steel
404-187819	Fitting - Brake master cylinder actuator link	95,000 to 135,000	MS 191*		Chrome-Moly Steel
404-187923	Fitting - Brake master cylinder actuator link	Forge per Army Spec. 57-105	MS 160 Type IV**		Mild Steel
404-188408	Fitting - Landing gear torque knee		14 STAL	Fed. QQ-A-367 Class 5	Aluminum Alloy
404-188410	Fitting - Landing gear shock absorber torque knee and drag leg	Normalized	MS 191*		Chrome-Moly Steel
404-188411	Fitting - Landing gear shock absorber fork and lower torque knee	Normalized	MS 262***		Chrome-Moly Steel
404-188426	Fitting - Landing gear axle bearing	Normalized	MS 191* or E4140	AN-QQ-S-752a	Chrome-Moly Steel
404-188441	Fitting - Landing gear axle bearing	Normalized	MS 191* or A8630		Chrome-Moly Steel
404-188446	Fitting - Landing gear rear leg	Normalized	MS 191* E4137 4140 A8630		Chrome-Moly Steel
404-188467	Fitting - Landing gear outer cylinder cap	95,000 to 135,000	MS 191*		Chrome-Moly Steel
404-188469	Fitting - Landing gear external stop	95,000 to 135,000	MS 191*		Chrome-Moly Steel
694-186094	Fitting - Stabilizer attach fitting	95,000 to 135,000	14ST Al	Fed. QQ-A-367 Class 5	Aluminum Alloy
804-188403	Fitting - Axle bearing	Normalized	MS 191*		Chrome-Moly Steel
804-188404	Fitting - Axle bearing cap	150,000	4130 carbon as high as .38%	MIL-S-6758	Chrome-Moly Steel
894-188565	Fitting - Landing gear retract mechanism Lower sprocket	Normalized	MS 191*		Chrome-Moly Steel
113674	Fitting - Safety belt fitting	180,000 to 200,000	MS 191*		Chrome-Moly Steel
114248	Fitting - Clevis	As forged	Commercial		
184217	Fitting - Wing root fitting	Normalized	4130	MIL-S-6758	Chrome-Moly Steel

TABLE OF HEAT TREATED FITTINGS (CONTINUED)

Part Number	Name	H.T., Lb/Sq. In.	Com'l Design	Specification	Material
186162	Fitting - Link end	Temper T4	14S-T4	Fed. QQ-A-367 Class 5	Aluminum Alloy
187321	Fitting - Aileron link blank		14ST	Fed. QQ-A-367 Class 5	Aluminum Alloy
187829	Fitting - Flap limit switch arm	95,000 to 135,000	NE8630		Chrome-Moly Steel
188101	Fitting - Landing gear V-brace	95,000 to 135,000	MS 191*		Chrome-Moly Steel
188103	Fitting - Landing gear slide	Normalized	MS 191*		Chrome-Moly Steel

*Parts made to Beech MS 191 may be made from any of the following steels:

4130H	MIL-S-6758
4140H	AN-QQ-S-752a
4135H	AN-QQ-S-686
8635H	
8735H	
8630H	

**Parts made to Beech MS 160 may be made from any of the following steels:

	Type IV
	A-1335
SAE & A	A1330
SAE	1320
SAE & C	1024
SAE & C	1025
SAE & C	1022
	1023
	1019
SAE & C	1020
	1018
SAE & C	1016

***Parts made to Beech MS 262 may be made from any of the following steels:

8635
8640
8735
8740
4140
4135

APPENDIX I

MATERIALS FOR REPAIR

STOCK DESCRIPTION	GAUGE	COMMERCIAL DESIGNATIONS	SPECIFICATION
Sheet	.020	24STAL Alclad	AN-A-13
Sheet	.025	24STAL Alclad	AN-A-13
Sheet	.032	24STAL Alclad	AN-A-13
Sheet	.040	24STAL Alclad	AN-A-13
Sheet	.020	52SH Alclad	Fed.QQ-A-318
Sheet	.025	52SH Alclad	Fed.QQ-A-318
Sheet	.032	52SH Alclad	Fed.QQ-A-318
Sheet	.040	52SH Alclad	Fed.QQ-A-318
Sheet	.051	52SH Alclad	Fed.QQ-A-318
Sheet	.032	52SO Alclad	Fed.QQ-A-318
Sheet	.032	17STAL Alclad	Fed.QQ-A-353
Sheet	.016	18-8 Type 802 Steel	AN-QQ-S-772
	101641	24ST	Fed.QQ-A-354
	K-11622	24ST Alclad	AN-A-13
	K-1508	24ST Alclad	AN-A-13
Strip	.064	17ST Alclad	Fed.QQ-A-316
Sheet		2SO Aluminum	QQ-A-561 Temper A
Extrusion	Bohn 86S4	17S Al Alloy	QQ-A-351
Extrusion	Alcoa T-533	61S Tubing	WW-T-789
Extrusion	Alcoa 772	24S Al Alloy	QQ-A-354
Extrusion	Alcoa 15514	24S Al Alloy	QQ-A-354
Extrusion	Alcoa D-17179	63ST Al Alloy	QQ-A-331
Extrusion	Alcoa D11623	24S Al Alloy	QQ-A-354
Extrusion	Alcoa D-17940	53S Al Alloy	QQ-A-331
Extrusion	Alcoa K-11622	24S Al Alloy	QQ-A-354
Extrusion	Alcoa K-16822	63S Al Alloy	QQ-A-331
Extrusion	Alcoa K-1508	24ST Al Alloy	QQ-A-354
Extrusion	Alcoa 875	24S Al Alloy	QQ-A-354
Extrusion	Alcoa 77S	17S Al Alloy	QQ-A-351

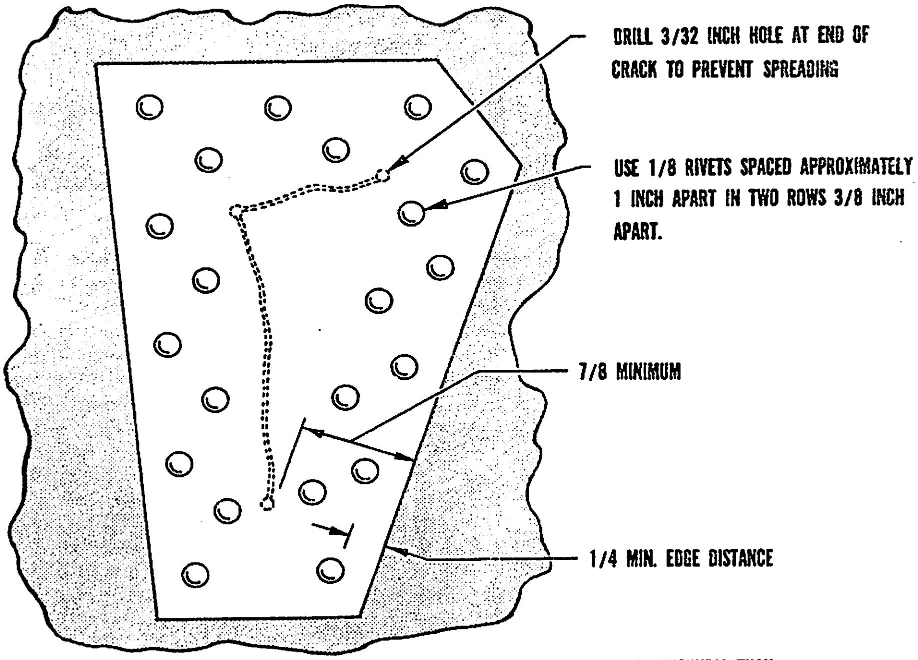
APPENDIX I

MATERIALS FOR REPAIR (CONTINUED)

STOCK DESCRIPTION	GAUGE	COMMERCIAL DESIGNATIONS	SPECIFICATION
Extrusion	Alcoa K-206B	17S Al Alloy	QQ-A-351
Extrusion	Reynolds E-1213	24S Al Alloy	QQ-A-351
Extrusion	Bohn 10013	24ST Al Alloy	QQ-A-354
Extrusion	Alcoa B10014	24ST Al Alloy	QQ-A-354
Extrusion		53S Al Alloy	QQ-A-331
Extrusion	Alcoa 734FF	24S Al Alloy	QQ-A-354
Extrusion	Alcoa 22088	24ST Al Alloy	QQ-A-331
Extrusion	Alcoa D20030	61S Al Alloy	QQ-A-331
Extrusion	Alcoa L23821	63S-6 Al Alloy	QQ-A-331
Extrusion	Alcoa D-20720	63S-6 Al Alloy	QQ-A-331
Extrusion	Alcoa K-22783	24S Al Alloy	QQ-A-354
Extrusion	Alcoa K-22784	24ST Al Alloy	QQ-A-354
802 Cotton Army Duck			CCC-D-771, Type III
Fabric, Grade A Flightex			AN-CCC-C-399
Thread, Sewing Machine Size 16/4 to 20/4 ply			Fed. V-T-276, Type IBI
Thread, Hand Sewing Size 8/4 Ply			Fed. V-T-276, Type IIIB
Cord, Lacing			MS-234
Tape, Reinforcing, 1/4-in.			AN-DDD-T-91a
Tape, Surface, 3-in.			6-62
Dope, Clear			AN-TT-D-514
Paste, Aluminum			Fed. TT-A-461
Dope, Aluminum			AN-TT-D-551
Thinner			AN-TTT-256
Thinner, Blush Retarding			AN-TTT-258

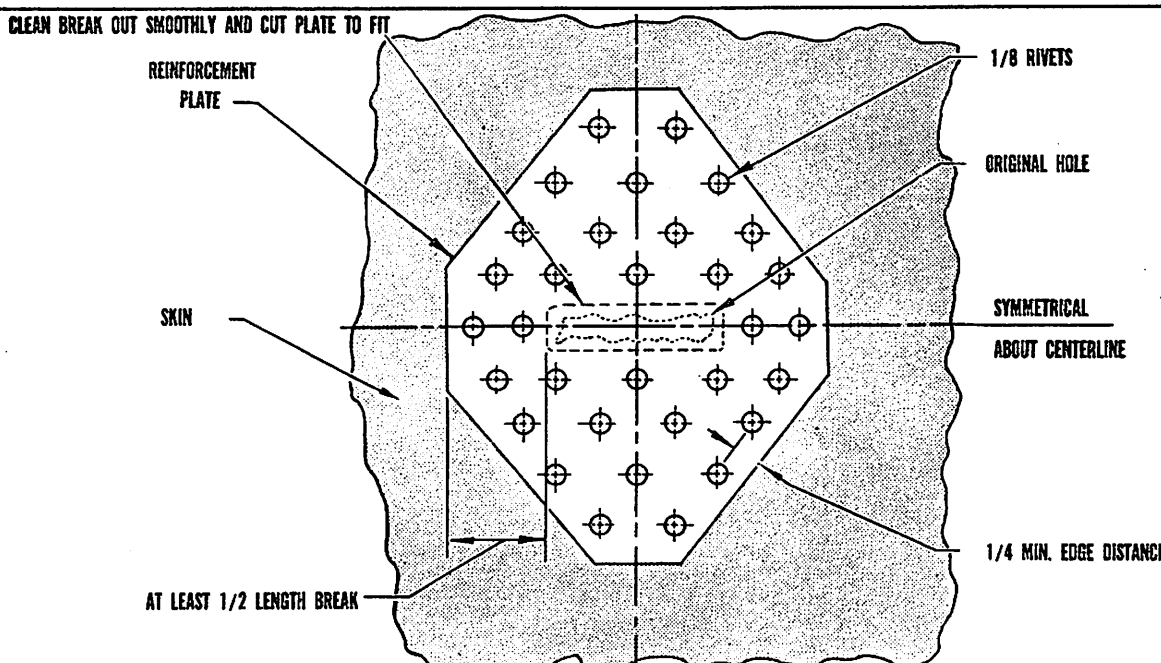
APPENDIX II

TYPICAL REPAIRS



NOTE — REINFORCEMENT PLATE MUST BE OF EQUAL OR GREATER THICKNESS THAN SKIN AND OF 17ST OR 24ST ALCLAD MATERIAL. USE A17ST RIVETS.

Figure B-1 Crack Repair



NOTE — REINFORCEMENT PLATE MUST BE OF EQUAL OR GREATER THICKNESS THAN SKIN AND OF 17ST OR 24ST ALCLAD MATERIAL. USE A17ST RIVETS.

Figure B-2 External Patch

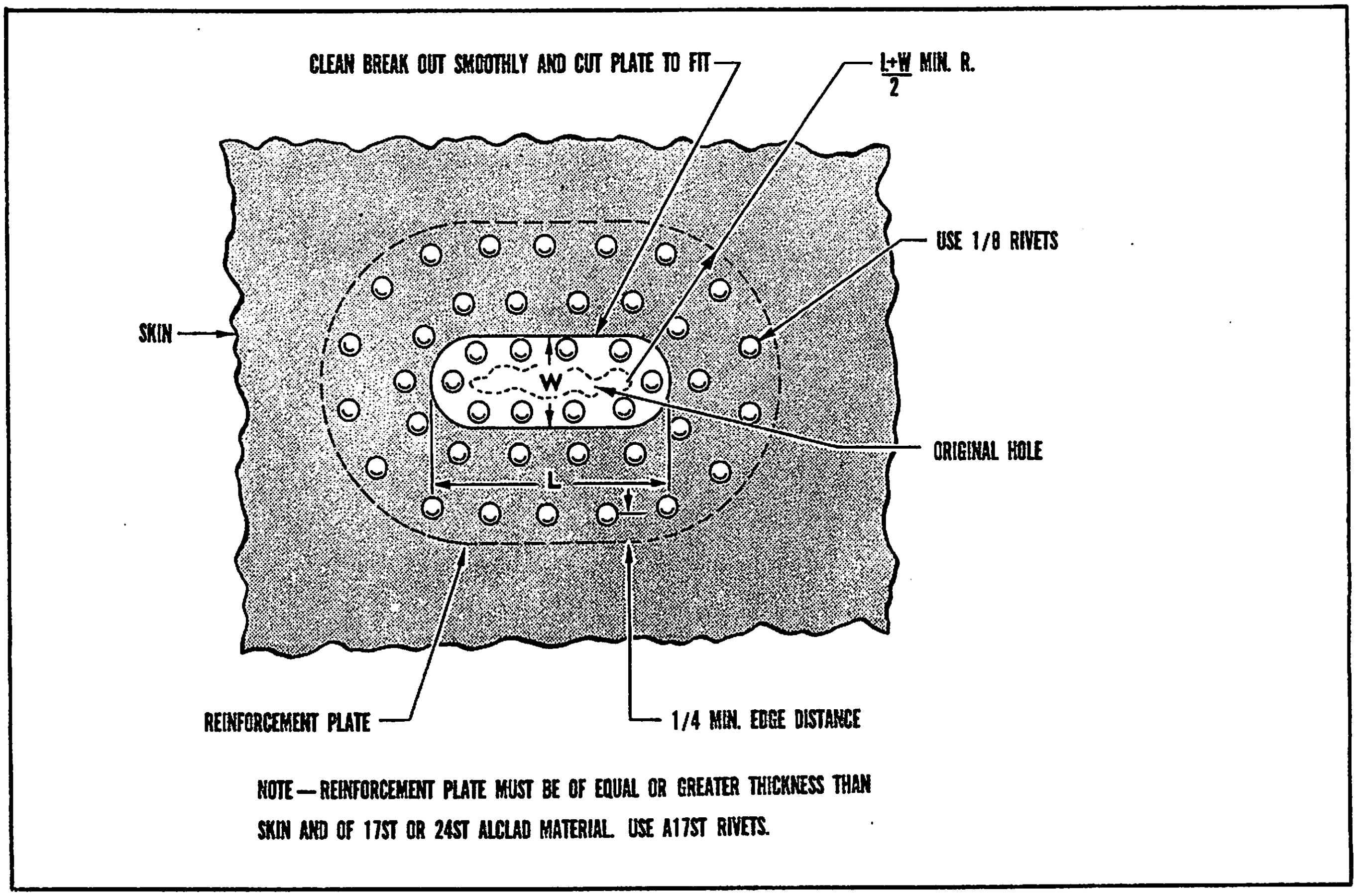


Figure B-3 Flush Patch

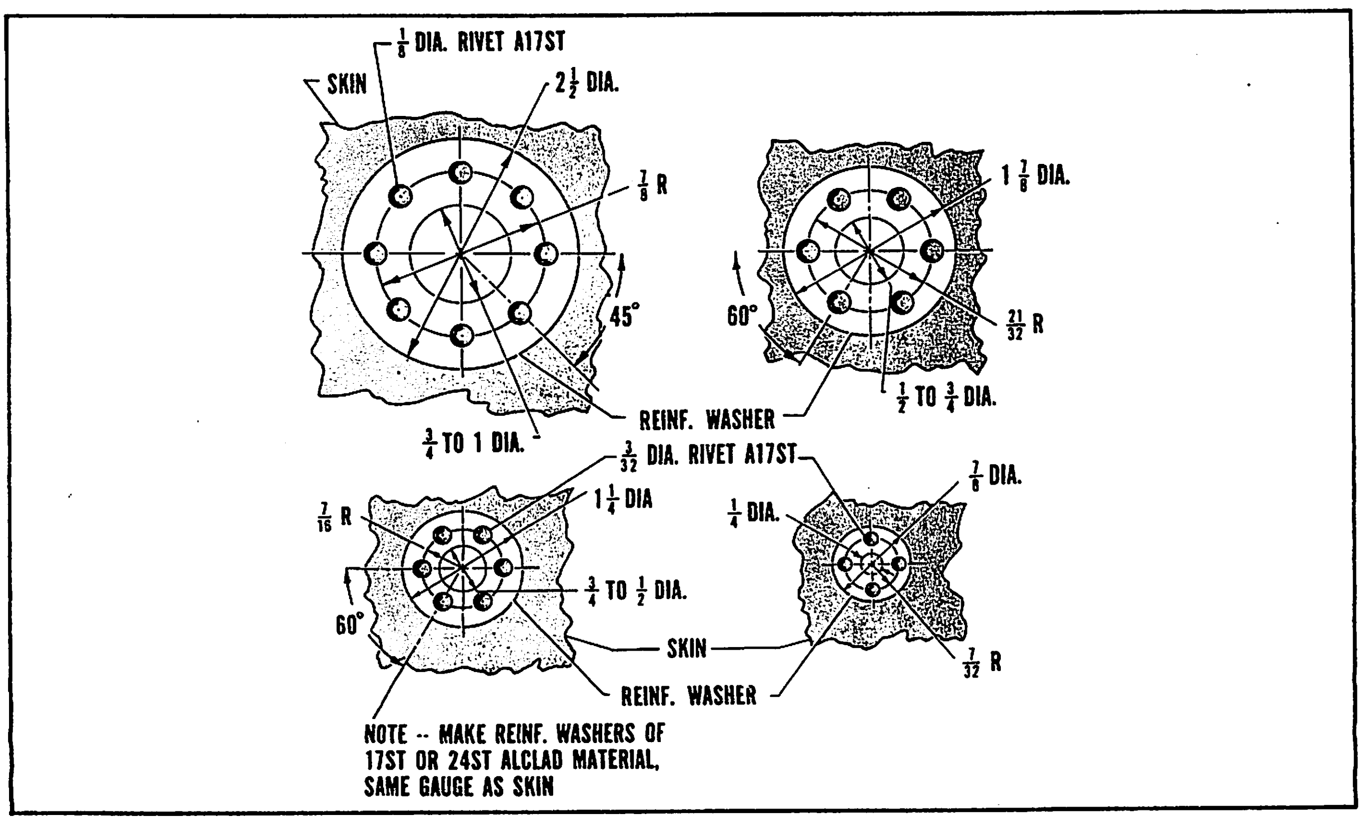


Figure B-4 External Circular Patch

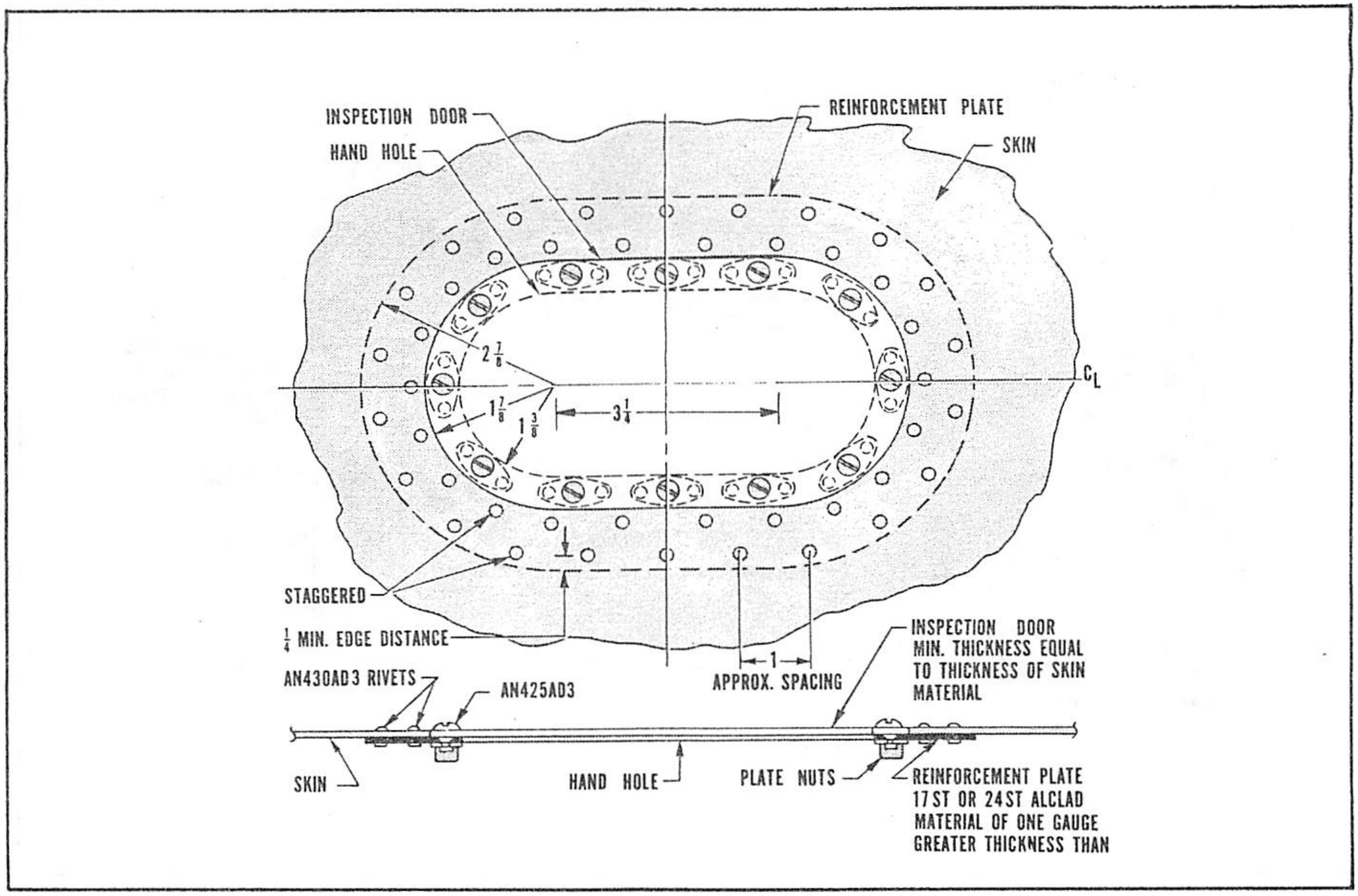


Figure B-5 Flush Access Hole

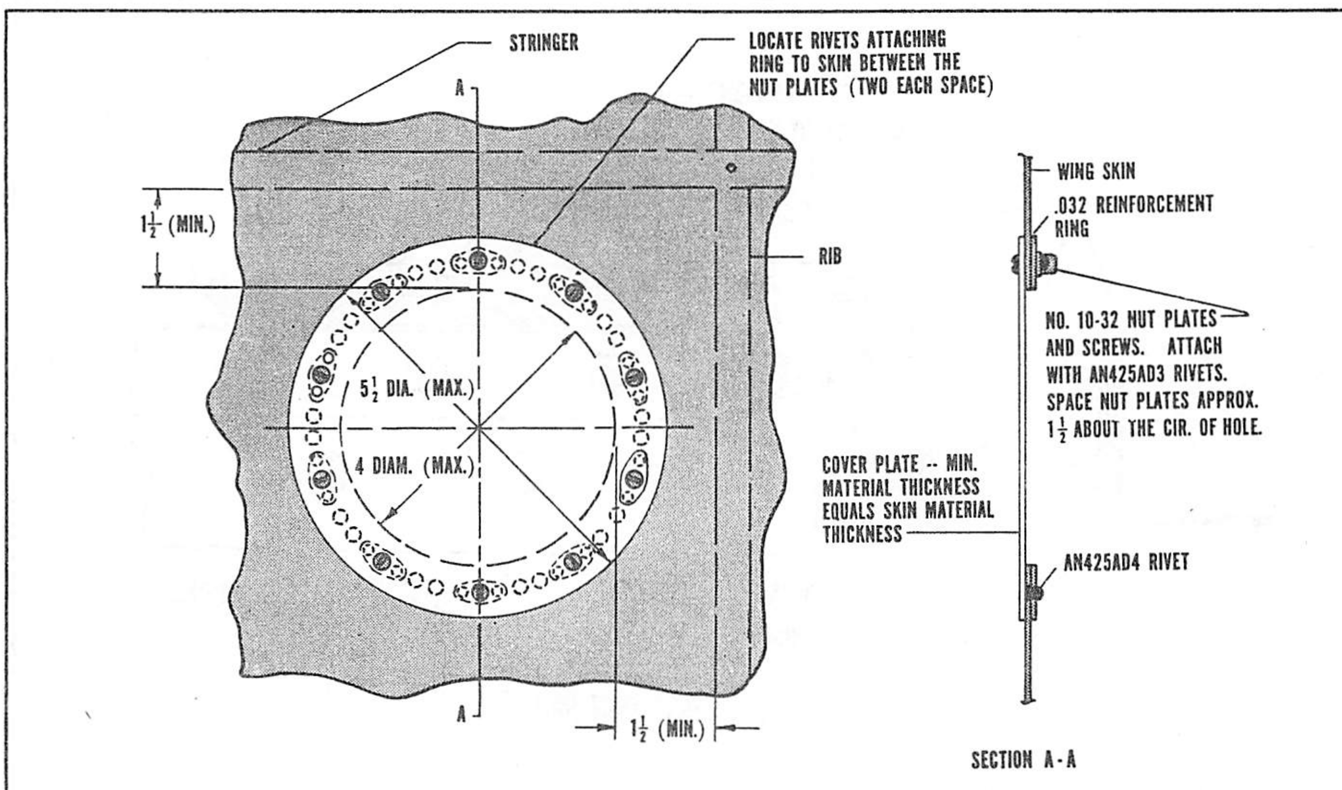


Figure B-6 External Access Hole

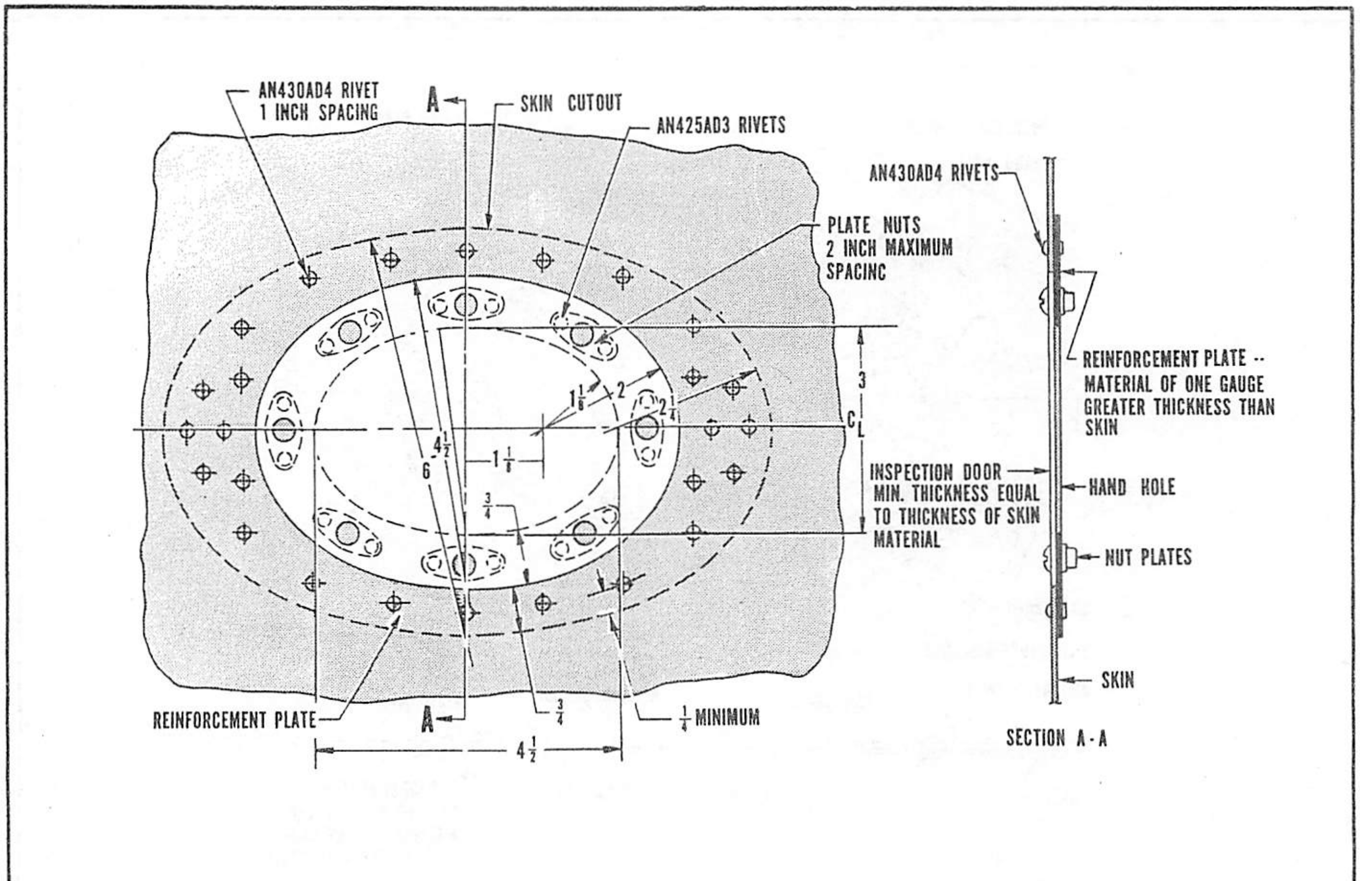


Figure B-7 Oval Flush Access

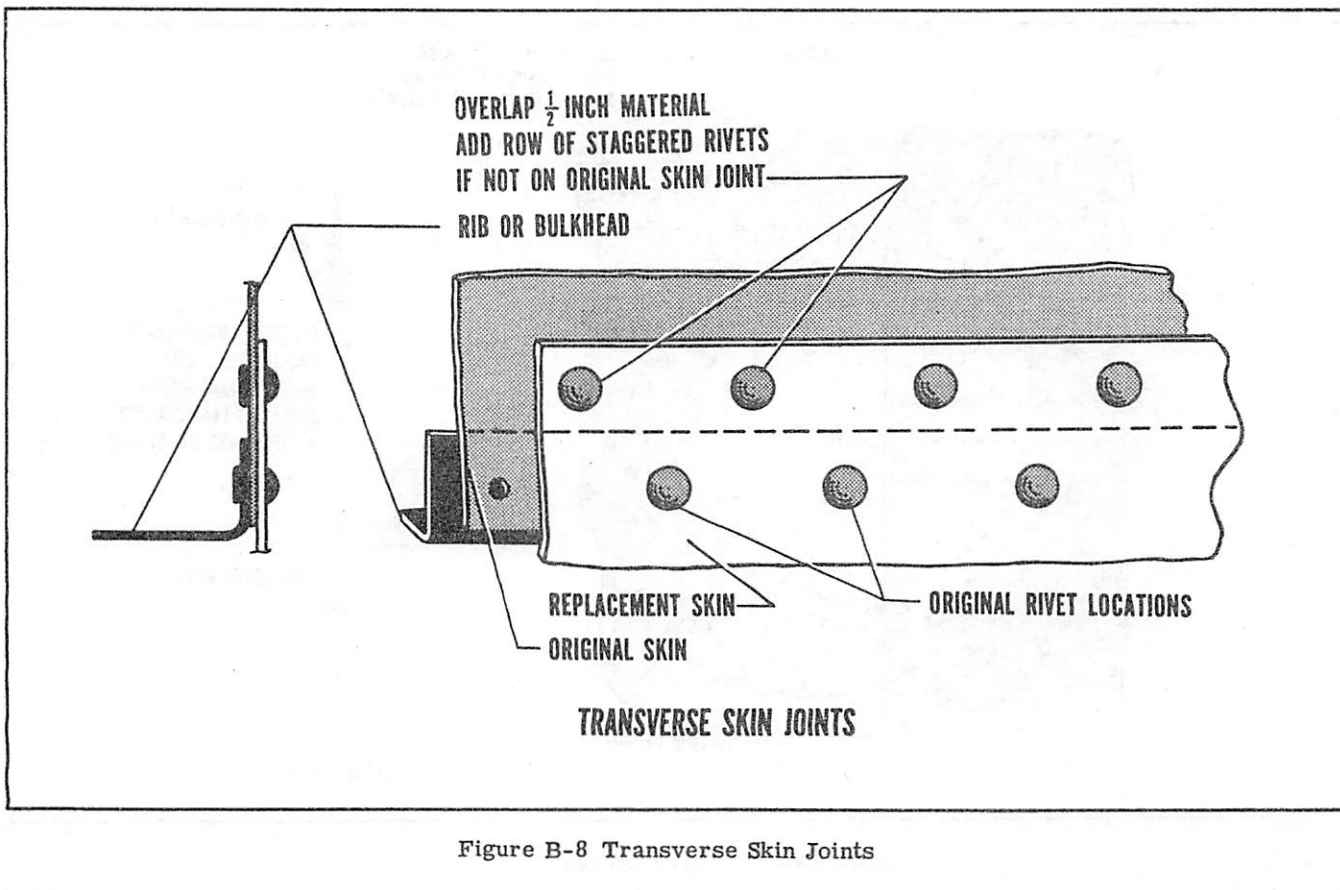


Figure B-8 Transverse Skin Joints

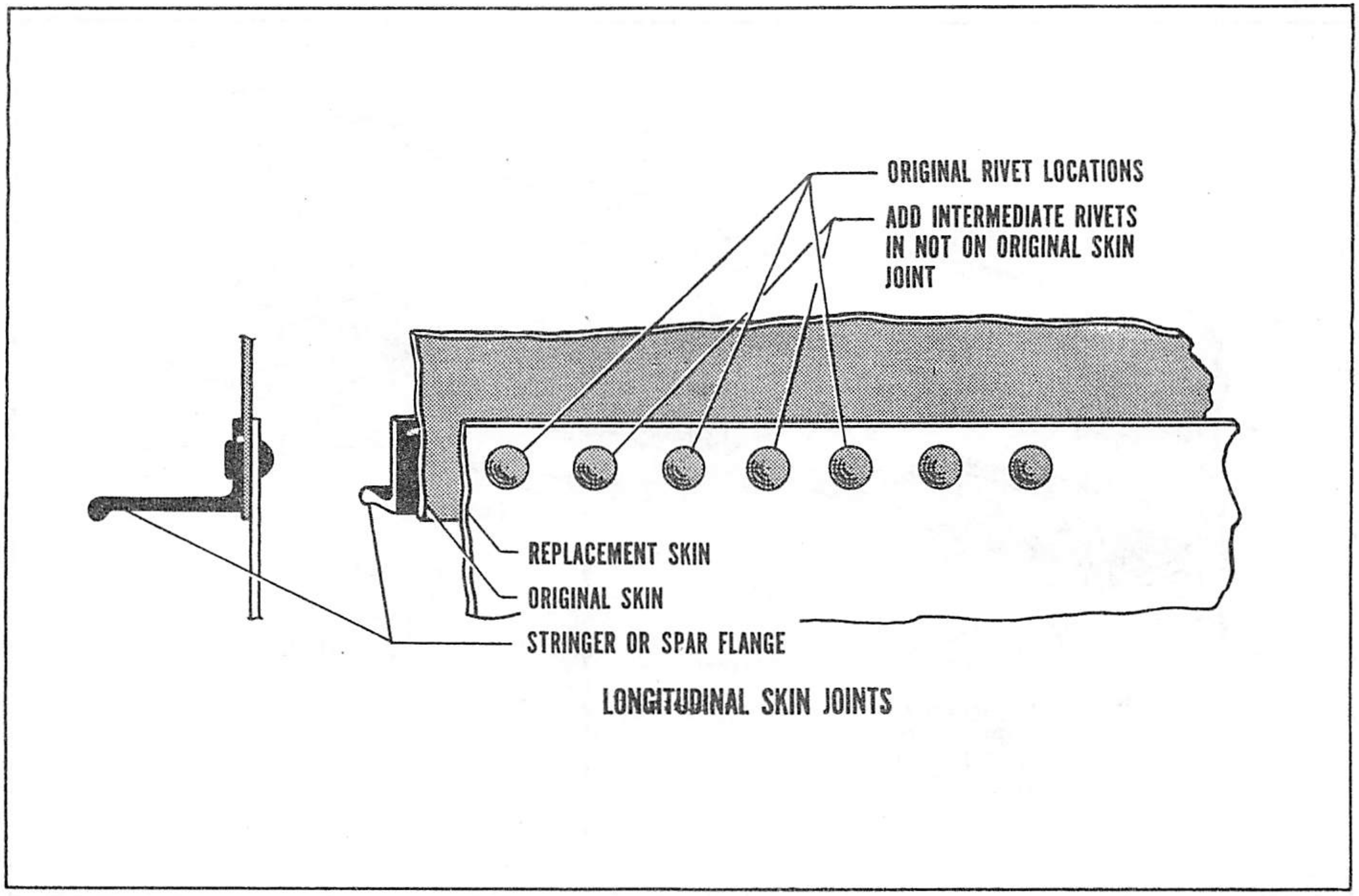


Figure B-9 Longitudinal Joint

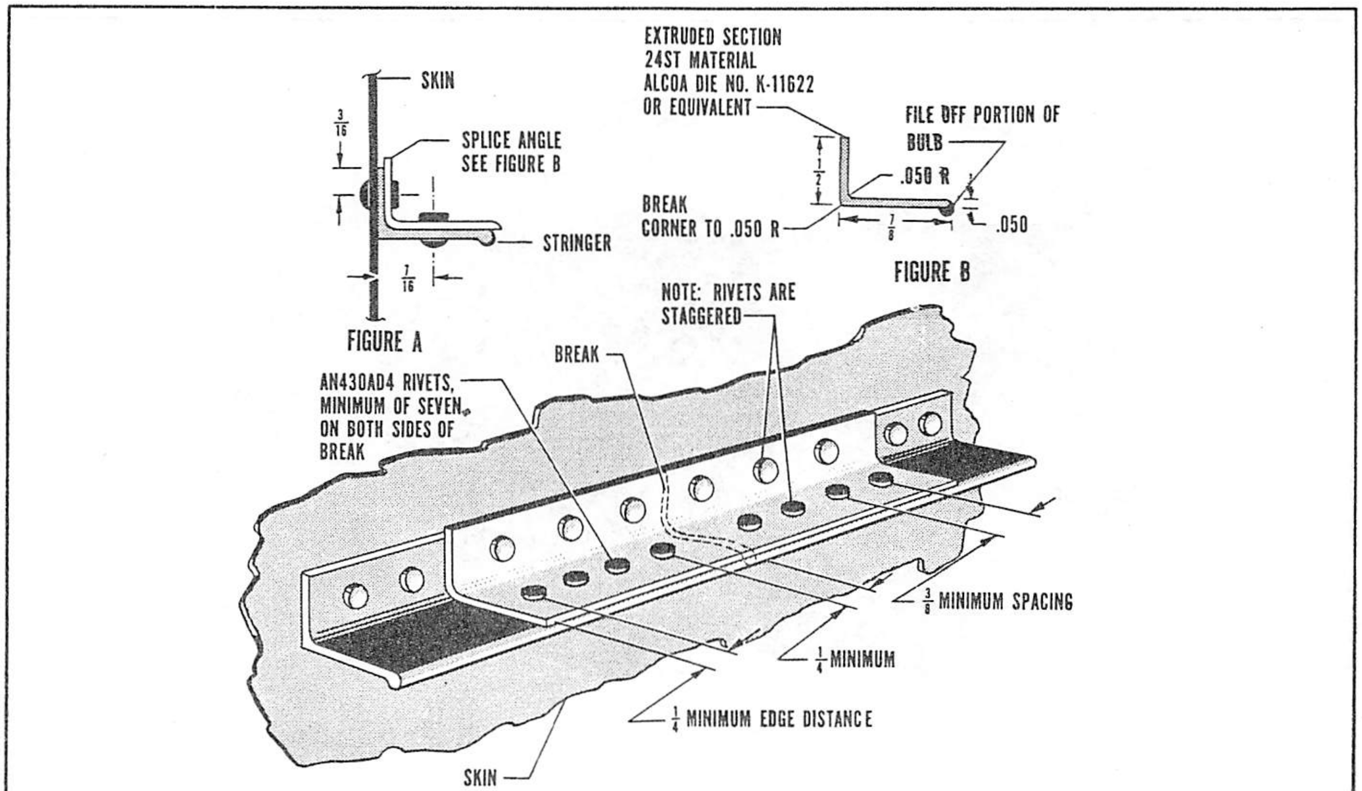


Figure B-10 Extruded Angle Stringer

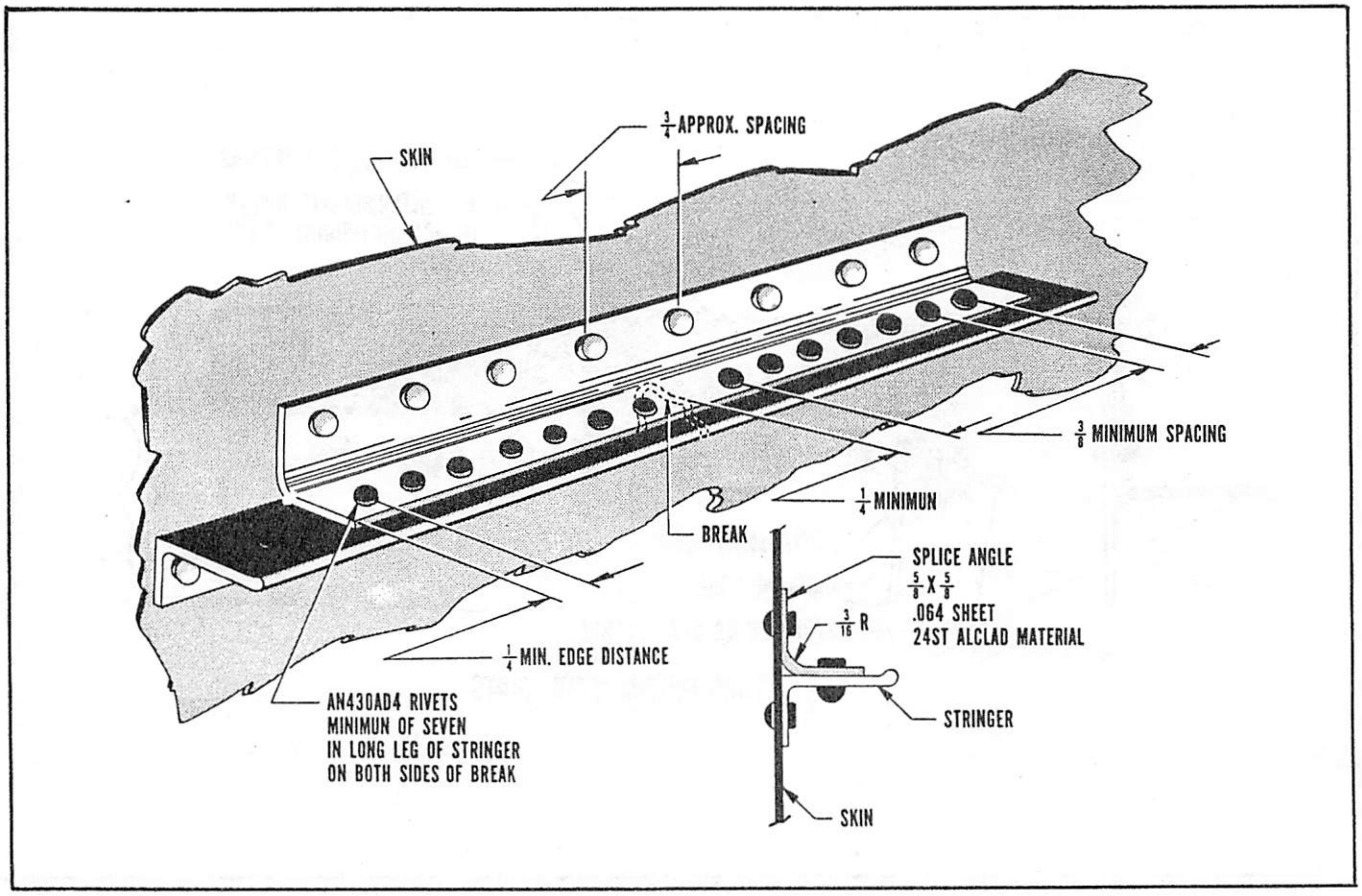


Figure B-11 Formed Stringer Repair

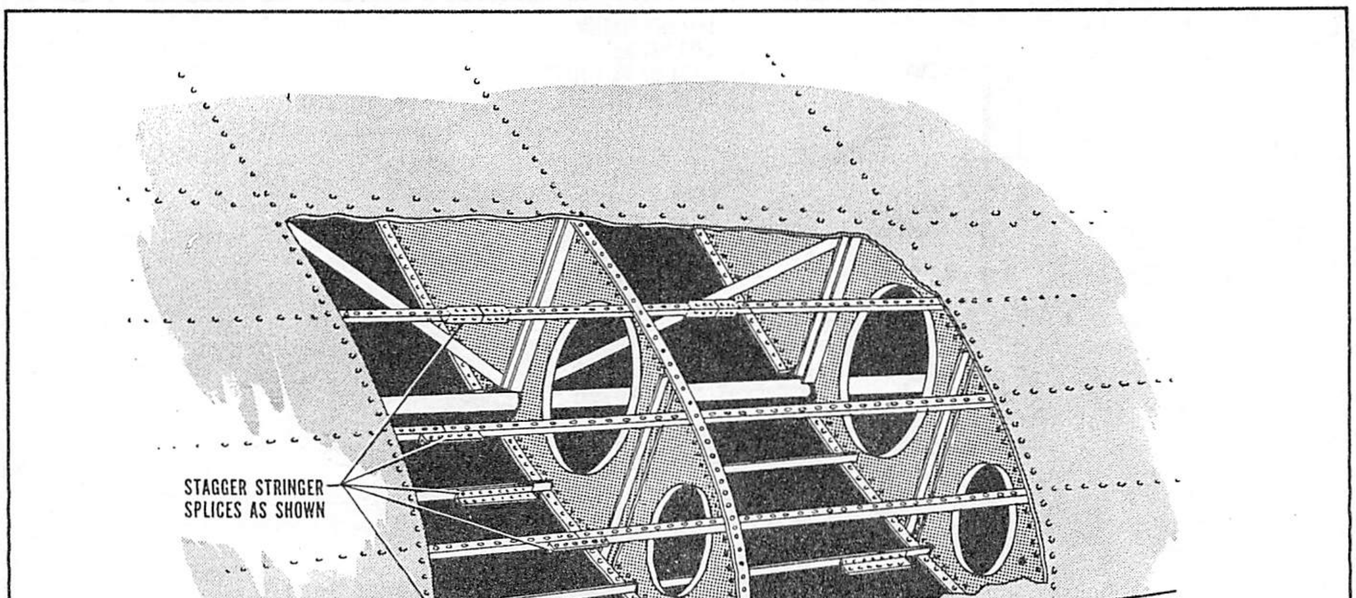


Figure B-12 Stringer Splices

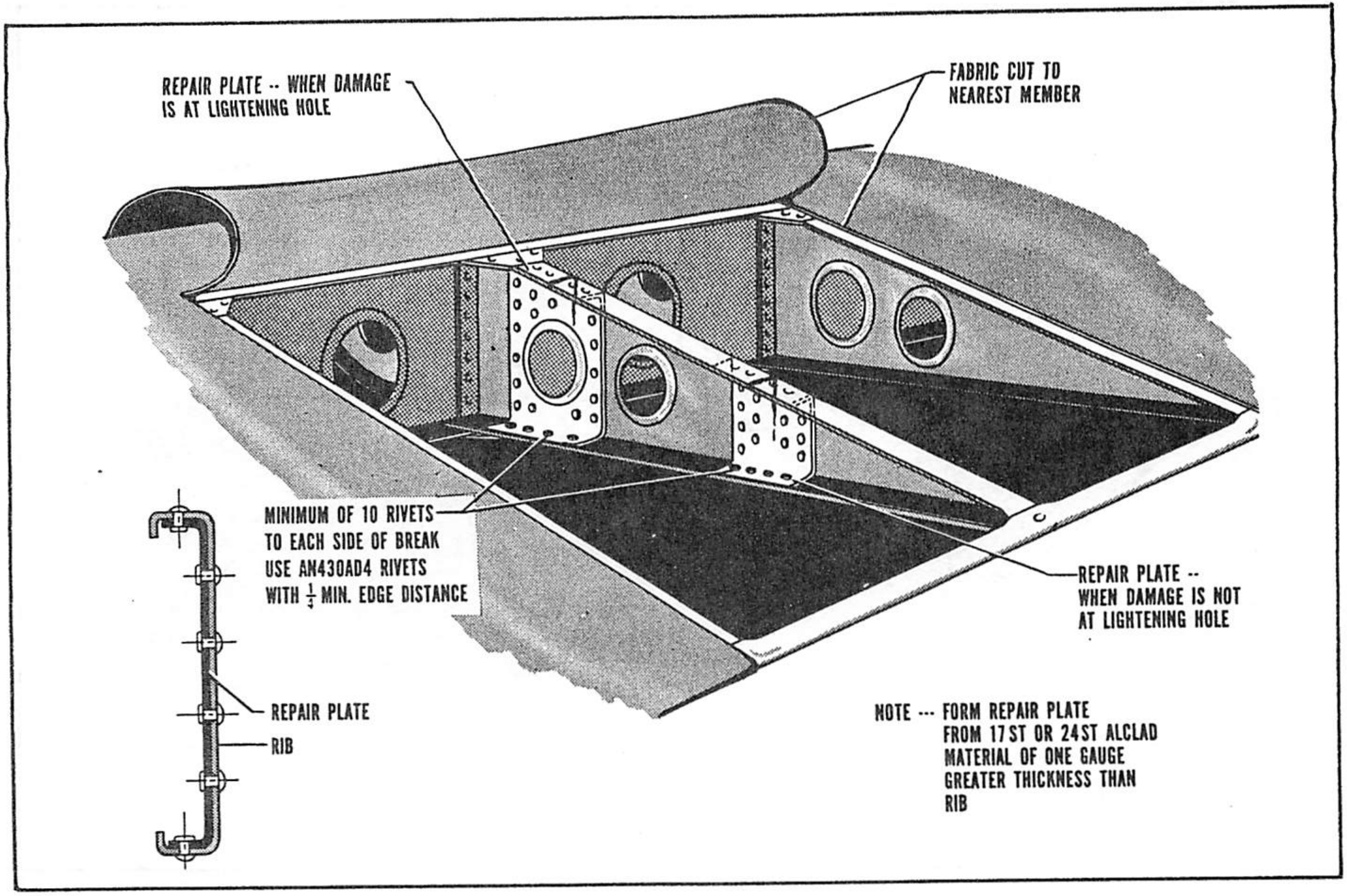


Figure B-13 Control Surface Rib

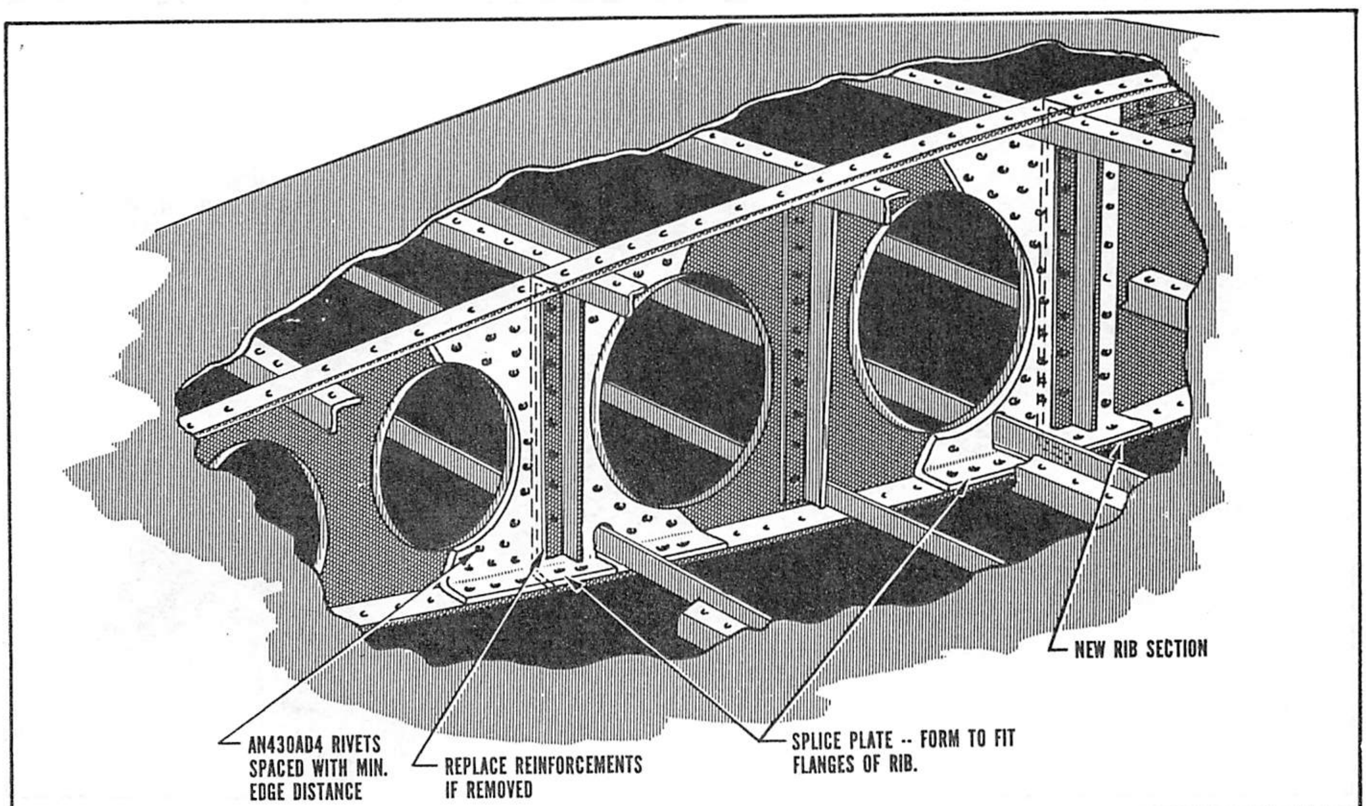


Figure B-14 Wing Rib

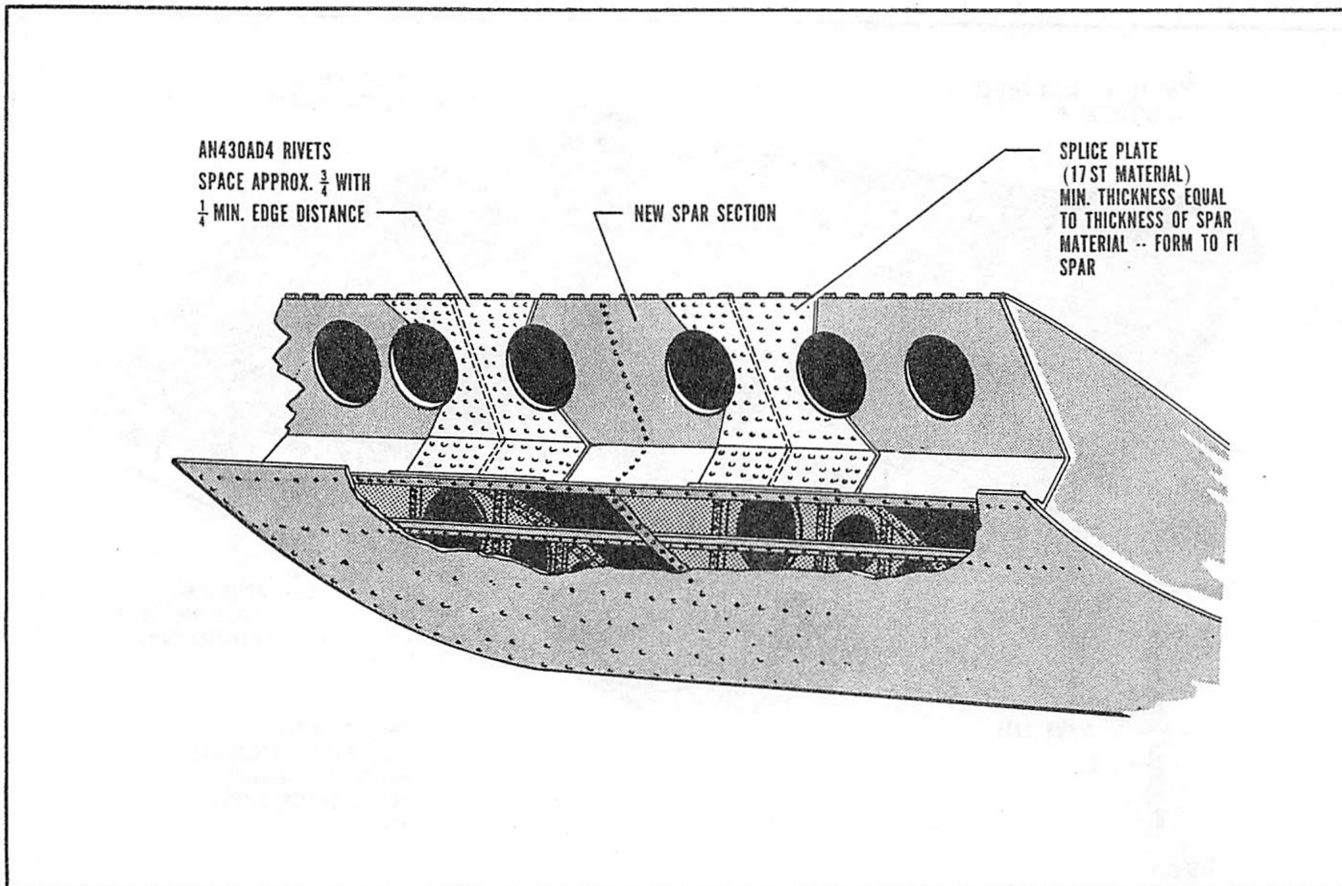


Figure B-15 Spar Repair

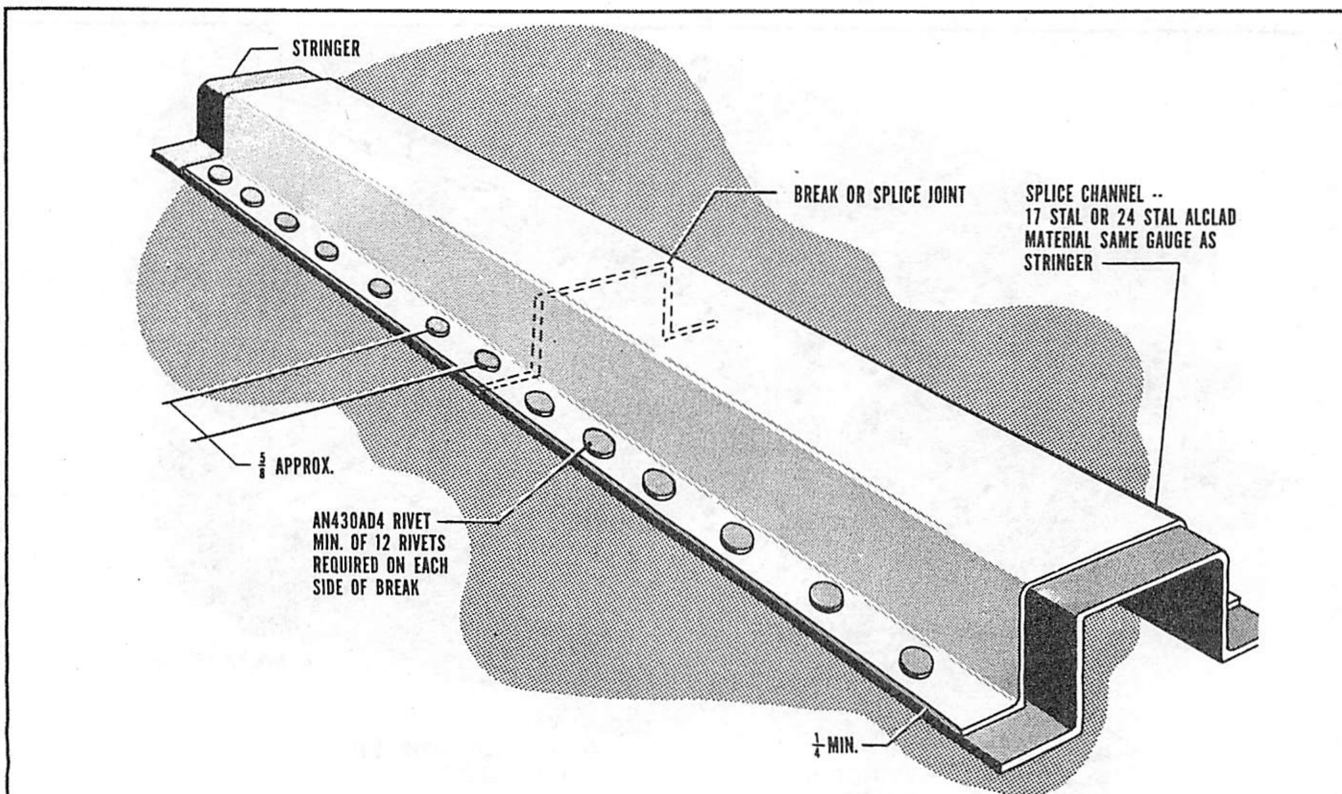


Figure B-16 Hat Section Repair

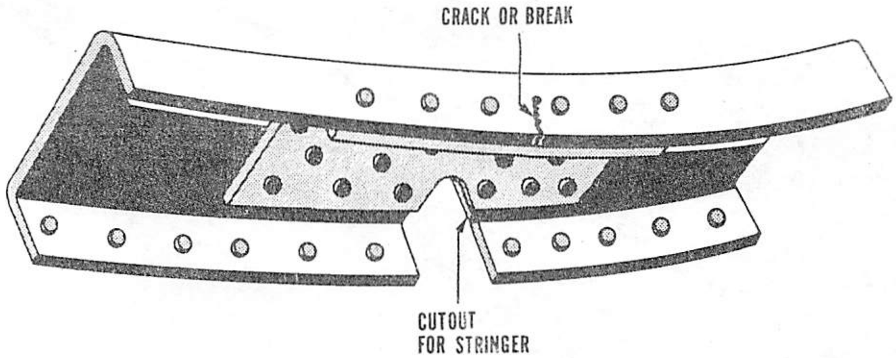


Figure B-17 Bulkhead Repair at Stringer Cutout

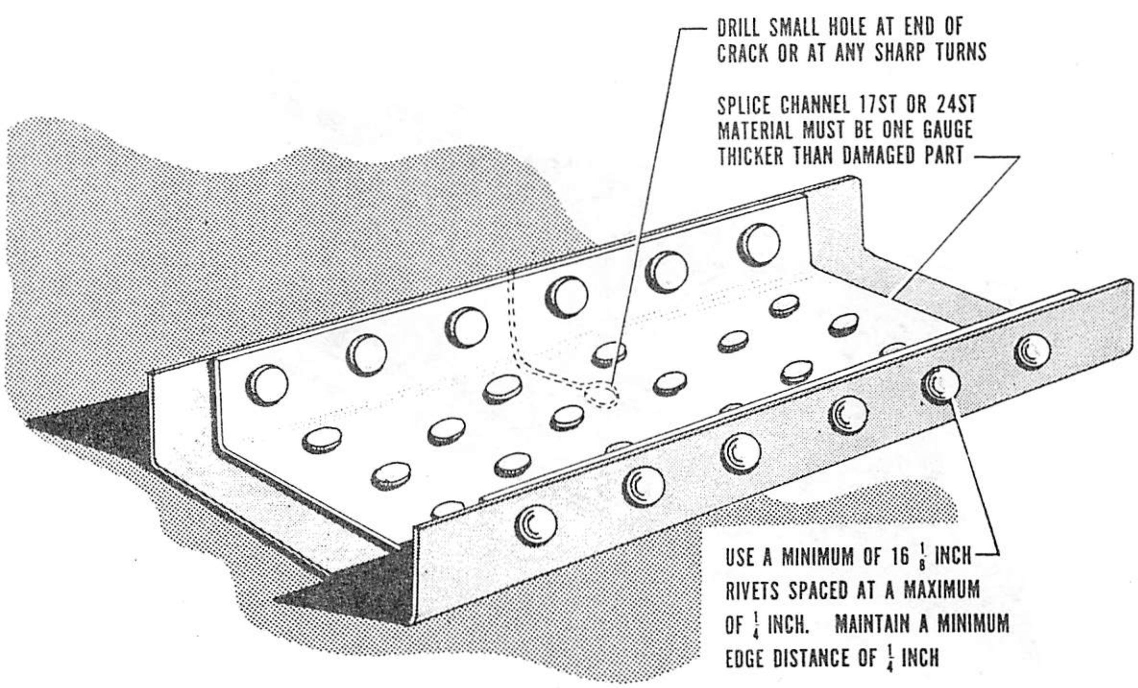


Figure B-18 Internal Splice, Bulkhead or U-Section

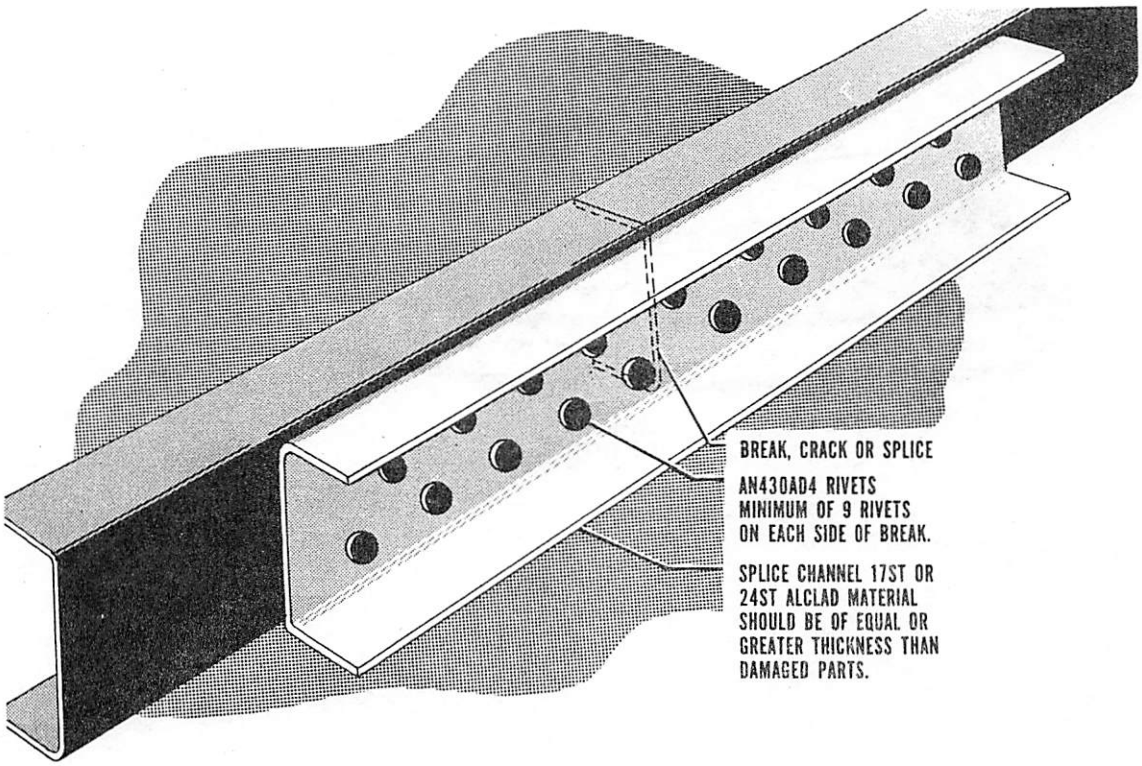


Figure B-19 External Splice, U-Section or Bulkhead

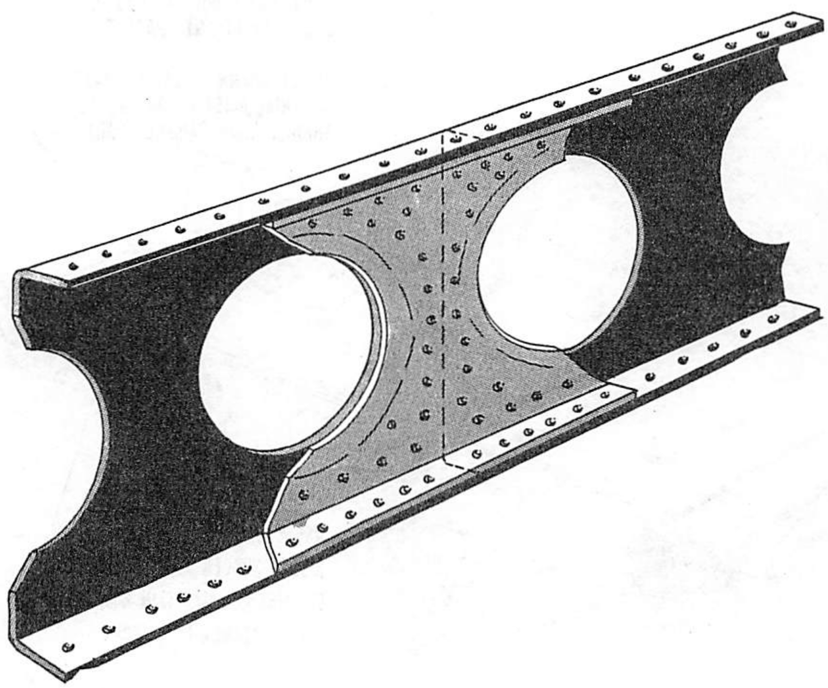


Figure B-20 Floor Channel

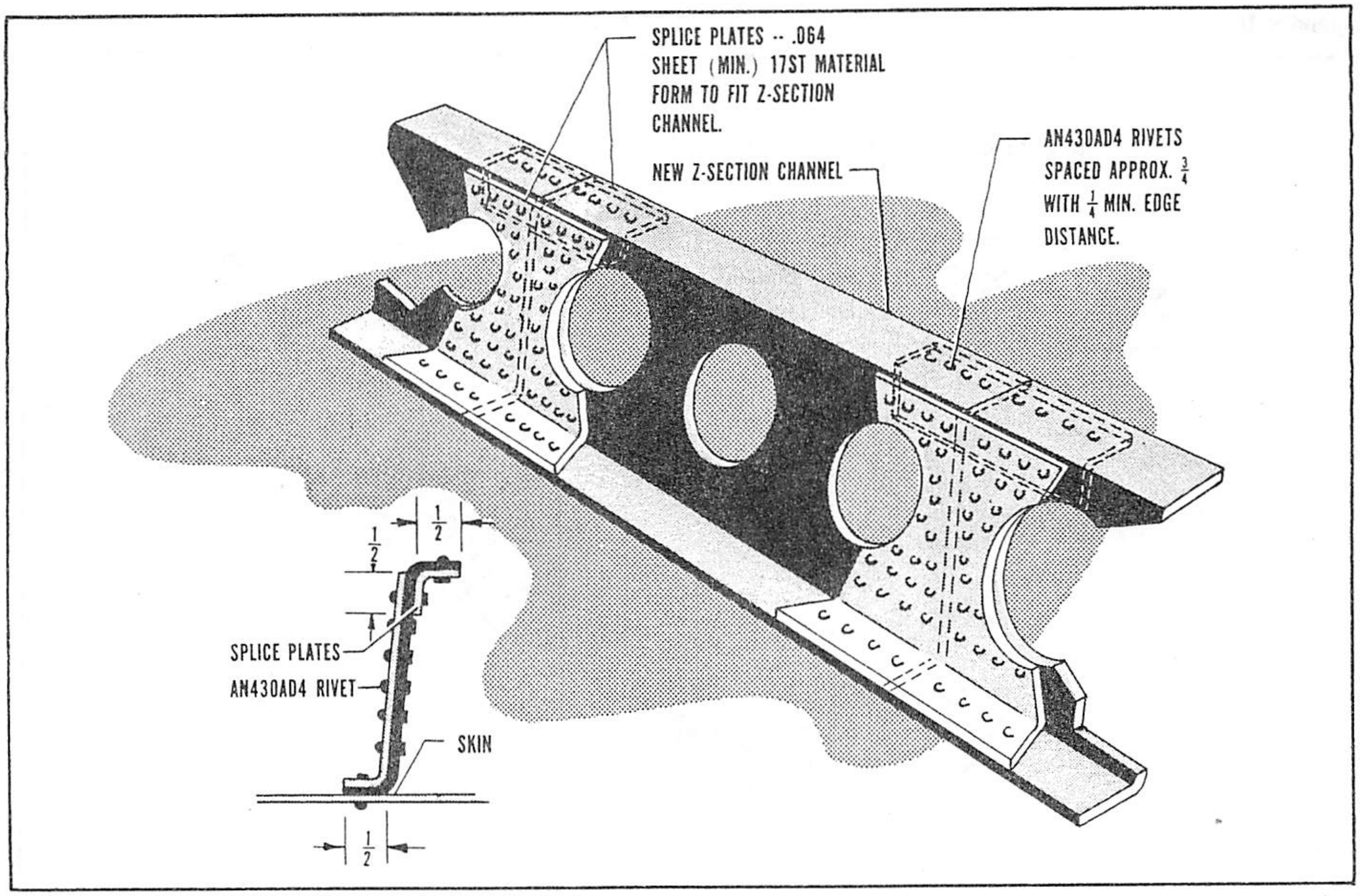


Figure B-21 Z- Channel Repair

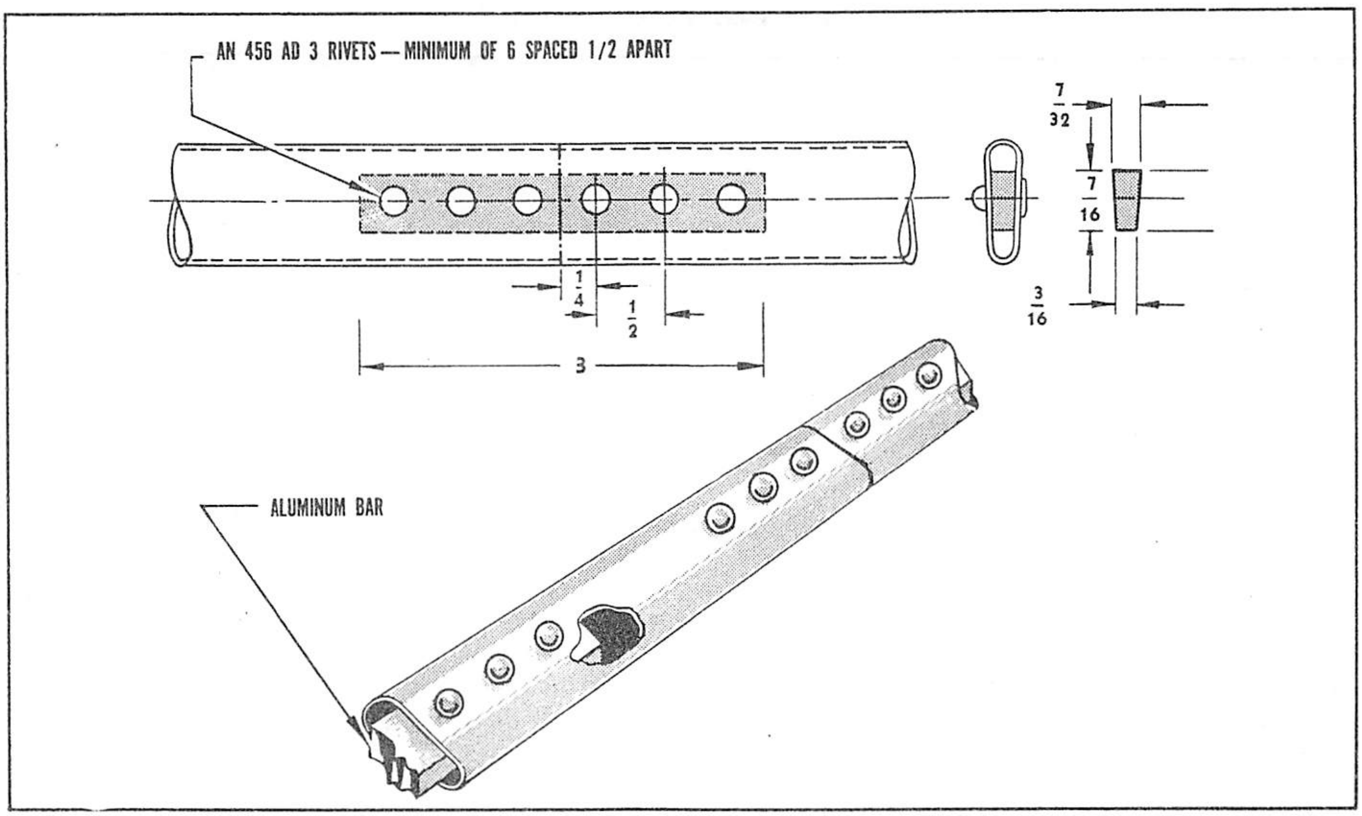


Figure B-22 Rudder Trailing Edge

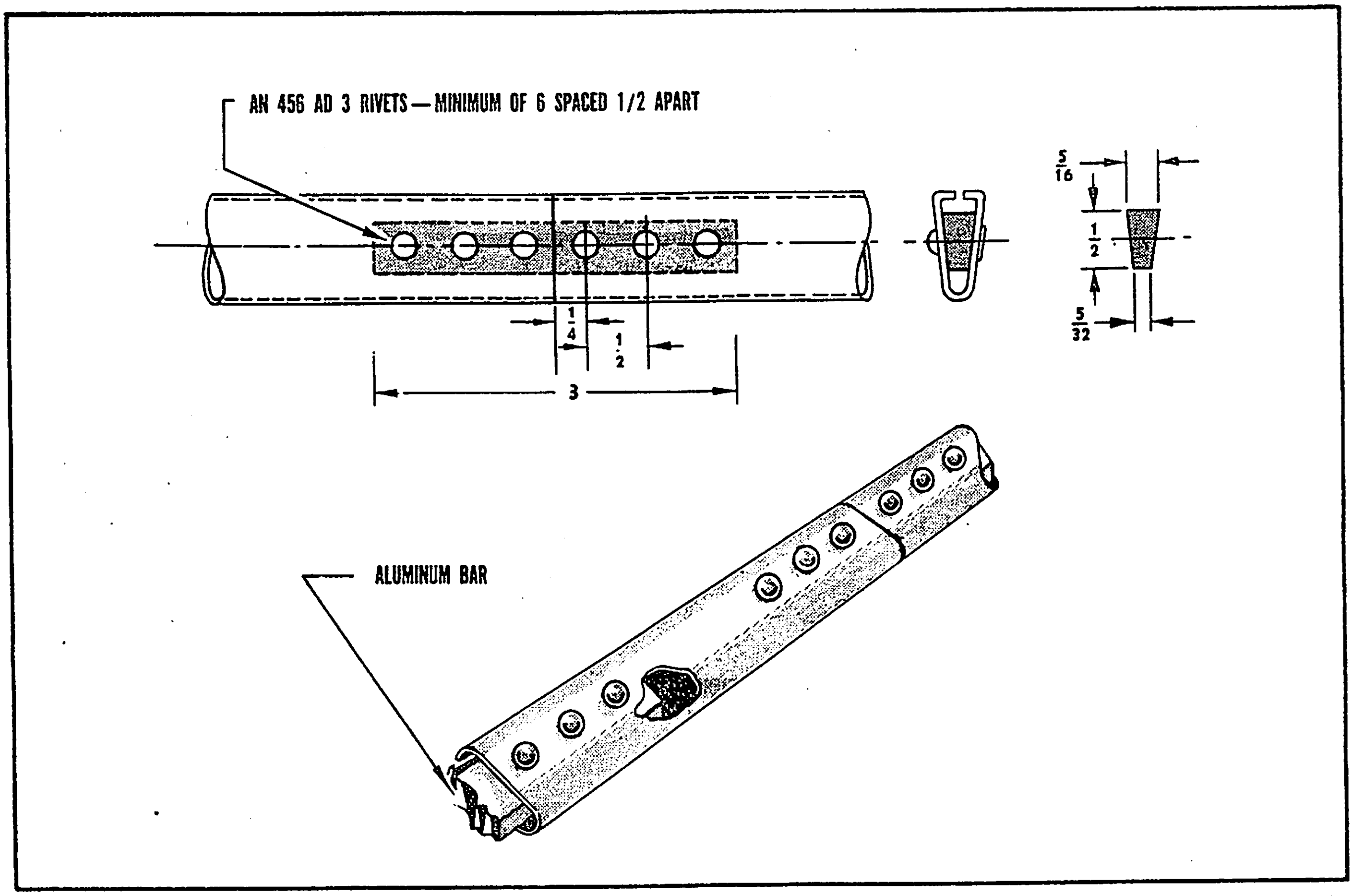


Figure B-23 Typical Trailing Edge