REPORT NO. NA-5614

Repair Manual

TRAINER AIRPLANES

SERIES

AT-6A, AT-6B, AT-6C
SNJ-3 and SNJ-4

NORTH AMERICAN AVIATION, INC.

DALLAS INGLEWOOD KANSAS CITY

TO THE HOLDER of THIS BOOK

If at any time contact cannot be made with the local N. A. A. representative, write to the field service department, NORTH AMERICAN AVIATION, INC., INGLEWOOD, CALIFORNIA, regarding the representatives location.

RESTRICTED

REPORT NO. NA-5614

Handbook of Instructions

for the

Structural Repair

of the "Texan"

TRAINER AIRPLANES

SERIES: AT-6A, AT-6B, AT-6C, SNJ-3 AND SNJ-4

CONTRACTS
AC-12969 CHANGE NO. 8
AC-15977
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NORTH AMERICAN AVIATION, INC.
DALLAS INGLEWOOD KANSAS CITY

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TABLE OF CONTENTS

SECTION 3
FIXED SURFACES

SECTION 4
MOVABLE SURFACES

SECTION 5
FUEL AND OIL SYSTEMS

SECTION 10
REBUSHING SPECIFICATIONS

SECTION 11
FINISH SPECIFICATION

LIST OF ILLUSTRATIONS

SECTION II
FUSELAGE

SECTION 3 FIXED SURFACES

SECTION 4
MOVABLE SURFACES

SECTION 5
FUEL AND OIL SYSTEM

TO THE REPAIRMAN:

In the preparation of this Manual, an attempt has been made to include extensive and complete information for the repair of damaged structural and nonstructural parts of the airplane. It is intended that this Repair Manual will facilitate the restoration of the airplane to active service within a minimum of elapsed time and with a minimum of replacement parts. If spare or replacement parts are required, they may be ordered as outlined below.

The repairs outlined in this Manual provide for the repair of any damage to any part of the aircraft and are not based upon any preconceived idea as to the extent of damage. Since the scope of possible damage is so broad it would be impossible to outline any exact procedure, only general repair methods are described and illustrated. These methods may have to be supplemented and altered to suit each specific situation. The material requirements, rivet specifications, etc., are the minimum allowable; and for the repair of any important structural components, personnel are warned against the use of any method of repair not described herein.

A complete explanation of each repair is outlined in the text. The illustrations should not be construed as being self-explanatory. The scheme of arrangement of each section is similar, dealing first with the description, then with repair procedures, materials, and tools required for making the various repairs. A thumb index has been inserted at the outside top of each page for quick identification of the section title and paragraph number.

For convenience, all common U. S. material specifications together with the British equivalents of these material specifications are listed at the end of Section I.

Your cooperation in reporting your views on this Manual and the material contained herein will help substantially in the further development of a more adequate manual. Opinions on the matter are solicited and may be communicated to the company through the North American Service Representative at your field or sent directly to the Field Service Department at Inglewood, California.

Field Service Department
North American Aviation, Inc.

SPARE PARTS

In many instances, incorrect part numbers, lack of complete information, and lack of part identification may cause delay in the filling of orders for spare parts. Requests by the company for additional information, necessitated by these discrepancies, involve loss of time; and the need for such can be eliminated if the correct part number, name, and location are clearly set forth in the original request. To facilitate the prompt fulfillment of requests, observe the following rules:

(1) Give correct part number and full title. *(Refer to the applicable structures diagram in this Manual or to the Parts Catalog, T.O. No. 01-60FE-4.)*

Example: 54-14015 left nose rib, station 15.087 (left wing, AT-6B)
88-14015-1 right nose rib, station 15.087 (right wing, AT-6C)
Note the prefix change for the AT-6C and that the part number ends with -1 signifying a right-hand part.

(2) Give the serial number and model of the airplane for which the part is intended.

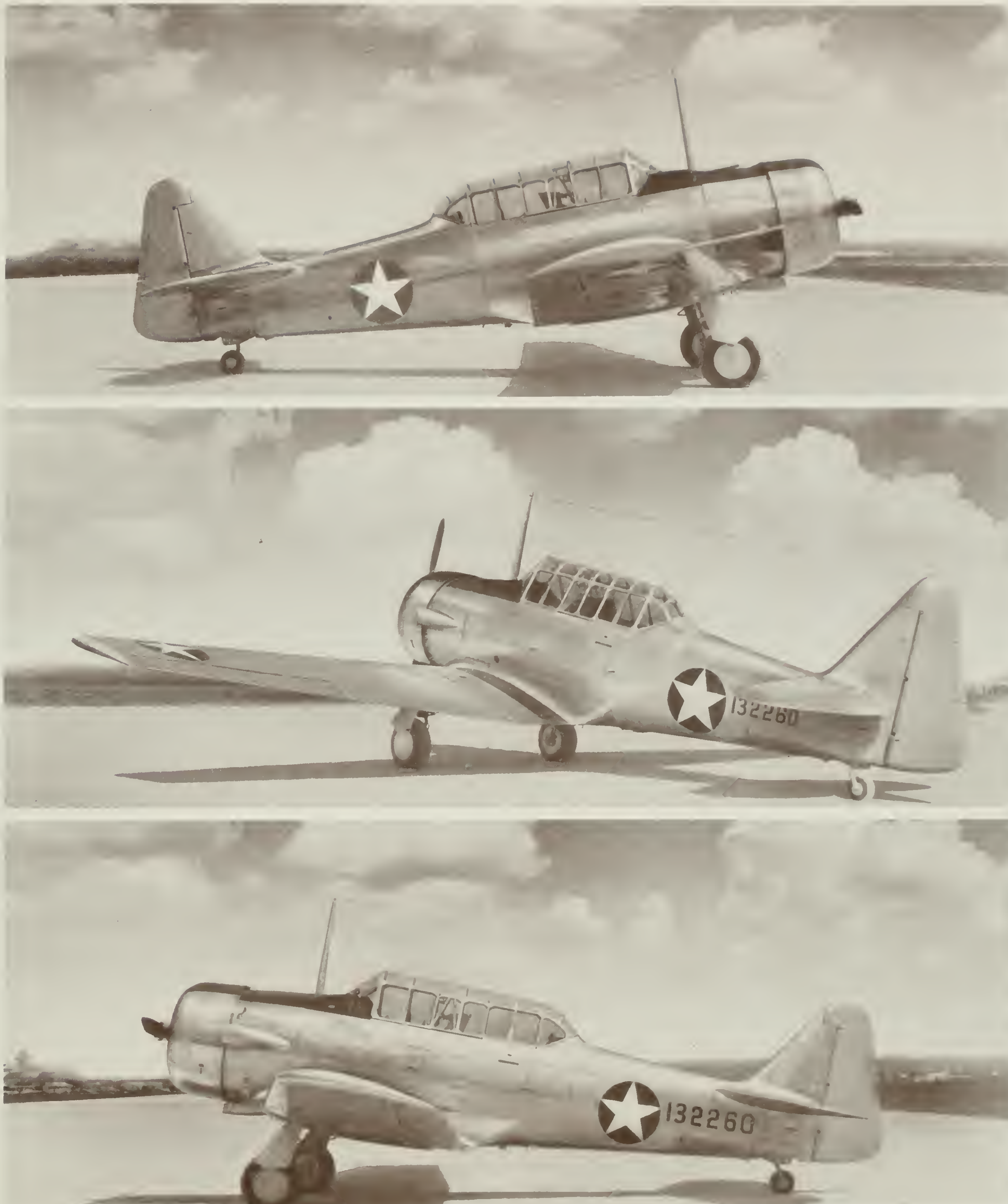
Example: Serial No. AC41-32218, Model AT-6C (This information is stenciled in the upper left-hand corner on the left-hand front fuselage side panel and stamped on a name plate on the right-hand side of the front cockpit.)

(3) If the part number or name is not available, furnish the location and complete description of the part and the unit or system in which the part will be used.

Example: The second nose rib outboard of the bolting angle of the left-hand wing assembly, AT-6C, Serial No. AC41-32218.

(4) Give complete dash numbers and title of all AN, AC, B, C, and NAF Standard Parts.

Example: AN3-4A bolt, AC365-832 nut, B1289-10-10 screw, C384-3-5 neoprene extrusion, NAF310621-S21D-4LH cable terminal.



Three Views of Complete Airplane

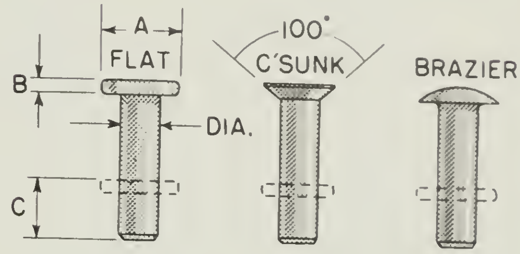
SECTION I
GENERAL INSTRUCTIONS

1. AIRPLANE CONSTRUCTION.

The AT-6 Series Airplane is a single-engine, low-wing, two-place advanced trainer. General airplane data are tabulated in this Section. *(See Figures 2 and 3.)* The airplane is equipped with a Pratt and Whitney, nine-cylinder, radial, air-cooled engine on which is mounted a Hamilton Standard, controllable pitch, constant speed, two-blade propeller. The main landing gear is fully retractable and the nonretractable tail wheel is full-pivoting. With the exception of the chrome molybdenum steel tubing front fuselage truss and engine mount, the airplane is primarily constructed of 24ST alclad sheet and 24ST aluminum alloy extruded members. The front and rear fuselage sections, the outer wing panels, and the empennage are constructed to facilitate removal and replacement in the event of damage. All structural members are readily accessible and are fabricated in a manner as to provide for ease of repair. It is to be noted that multiplicity of types and sizes of component parts is confined to an absolute minimum in original construction of the airplane and also in repair. North American Aviation, Inc., major component part numbers are listed and illustrated in this Section. *(See Figure 4.)*

2. RIVET TYPES.

There are three different types of rivets used in the construction of the airplane *(see Figure 1)*. All are made of A17S aluminum alloy material (rivet designation AD), and thus no heat treatment is required prior to driving. Rivets of A17S material are marked with an identifying dent or depression on the center of the manufactured rivet head. These three types of rivets employed in the construction of the airplane are as follows: (1) the Type AN442-AD flat head rivet, used for the assembly of all interior structure; (2) the Type B1219-AD (superseded by Type AN426-AD for all future procurement) 100-degree countersunk head rivet, used to reduce drag on the forward skin of the fuselage and fixed surfaces; (3) the Type B1227-AD brazier head rivet, used on the aft skin of the fixed surfaces and fuselage. In making repairs, these types of rivets should be used in their original function. The rivet coding used throughout this Manual specifies the type of rivet and the size. For example, a notation "AN442-AD5 rivet" signifies a flat head rivet of A17S material. The last number of the rivet notation always signifies the rivet diameter in 1/32-inch. Therefore, the rivet diameter in this case is 5/32-inch.



AN442-AD B1219-AD B1227-AD
(AN426-AD)

APPROXIMATE DIMENSIONS

DIA.	3/32	1/8	5/32	3/16
A	3/16 (.172*)	1/4 (.219*)	5/16 (.281*)	3/8 (.344*)
B	.038	.044	.055	.066
C	9/64	3/16	15/64	9/32

* B1219-AD ONLY

Figure 1—Types of Rivets

3. GENERAL RIVETING EQUIPMENT.

Various tools required in the normal course of riveting include twist drills, reamers, rivet cutters or nippers, pneumatic rivet guns, bucking bars, hammers, draw sets, rivet sets, dimple dies, and countersinking drills. Other accessories which are often particularly useful are machine screws, clamps, fasteners, and masking tape. Inasmuch as proper equipment is essential for good riveting, considerable attention should be given to the choice of riveting tools and method of riveting.

4. DRILLING.

Standard twist drills should be used to drill rivet holes. Locations for the rivet holes

TABLE OF SPECIFICATIONS

OVERALL WING SPAN	42 ft. 1/4 in.
OVERALL LENGTH	28 ft. 11-7/8 in.
OVERALL HEIGHT ANTENNA MAST, THRUST LINE LEVEL	12 ft. 9-1/4 in.
OVERALL HEIGHT ANTENNA MAST, THREE-POINT	12 ft. 6 in.
HEIGHT, PROPELLER HUB, THRUST LINE LEVEL	65-49/64 in.
CLEARANCE, PROPELLER TIPS, THRUST LINE LEVEL	11-49/64 in.
DIAMETER OF PROPELLER	9 ft.
NORMAL C.G. TO WING LEADING EDGE - WHEELS DOWN	30.54 in.
NORMAL C.G. TO WING LEADING EDGE - WHEELS UP	30.27 in.
C.G. LIMITS TO WING LEADING EDGE	29.60-33.75 in.
ELEVATOR HINGE TO WING LEADING EDGE	235-5/8 in.
MINIMUM CLEARANCE OF PROPELLER AND RING COWL	1 in.
ANGLE OF NORMAL C.G. WITH WHEEL *(Side Elevation)*	18 deg.,11 min.
ANGLE OF REARWARD C.G. WITH WHEEL *(Side Elevation)*	20 deg.,41 min.
ANGLE OF WING TIP AND WHEEL BOTTOM *(Front Elevation)*	16 deg.,30 min.
ANGLE OF THRUST LINE WITH GROUND *(3-Point Position)*	12 deg.,30 min.

LANDING GEAR

TREAD	102-3/4 in.
STRUT TILT, FORWARD OF VERTICAL	7 deg.13 min. 27 sec.
SIZE OF WHEELS, TIRES & TUBES *(Smooth Contour)*	27 in.
SIZE OF TAIL WHEEL TIRE *(Smooth Contour)*	12-1/2 in.

WING

AIRFOIL SECTION AT ROOT	NACA 2215
AIRFOIL SECTION AT TIP	NACA 4412
TOTAL AREA, INCL. AILERONS	253.73 sq.ft.
SPAN	42 ft. 1/4 in.
MEAN AERO. CHORD *(Semi-Wing)*	75.96 in.
LEADING EDGE OF M.A.C. *(Location behind L.E. Centersection)*	11.26 in.
CHORD AT ROOT	90 in.
CHORD AT TIP	48 in.
DIHEDRAL - LEADING EDGE	5 deg. 41 min.
INCIDENCE AT ROOT	2 deg.
INCIDENCE AT TIP	0 deg.
WASHOUT AT TIP	2 deg.
SWEEPBACK - L.E.	10 deg.,18 min.,16 sec.

AILERONS *(Each)*

AREA AFT OF HINGE LINE, *(Incl. Booster Tab)*	8.98 sq.ft.
AREA OF BALANCE	2.42 sq.ft.
TOTAL AREA OF EACH AILERON	11.40 sq.ft.
AREA OF BOOSTER TAB	.38 sq.ft.
AILERON TRAVEL	UP 30 deg. - DN 15 deg.
BOOSTER TAB TRAVEL	UP 15 deg. - DN 30 deg.

FLAPS

AREA OF CENTERSECTION FLAP	11.40 sq.ft.
AREA OF EACH OUTER WING FLAP	8.73 sq.ft.
TOTAL AREA *(3 Flaps)*	28.86 sq.ft.
CHORD, EACH FLAP	14-11/32 in.
LENGTH OF CENTERSECTION	113-3/4 in.
LENGTH OF EACH OUTER WING FLAP	89-3/8 in.

EMPENNAGE

TOTAL AREA HORIZONTAL TAIL SURFACE *(Incl. 5.33 sq.ft. Fuselage)*	50.11 sq.ft.
TOTAL AREA VERTICAL SURFACES	18.54 sq.ft.

HORIZONTAL STABILIZER

TOTAL AREA *(Incl. 5.33 sq.ft. Fuselage)*	28.34 sq.ft.
INCIDENCE *(fixed setting)*	1 deg.
OVERALL SPAN	12 ft. 11-5/16 in.
CHORD, MAX. *(L.E. to Hinge Line)*	38-7/8 in.

ELEVATORS *(Each)*

SPAN	69-21/32 in.
AREA AFT OF HINGE LINE *(Incl. Tab)*	9.13 sq.ft.
AREA TRIM TAB	.73 sq.ft.
TOTAL AREA *(Incl. Tab & Balance)*	10.88 sq.ft.
ELEVATOR TRAVEL	UP 30 deg. - DN 20 deg.
TRIM TAB TRAVEL	UP 8 deg. - DN 16 deg.

VERTICAL STABILIZER

AREA	5.33 sq.ft.
SETTING *(Fixed)*	1-3/4 deg. Left
CHORD,MAX. *(L.E. to Hinge Line)*	51-11/16 in.

RUDDER

AREA AFT OF HINGE LINE *(Incl. Tab)*	10.95 sq.ft.
AREA OF BALANCE	2.26 sq.ft.
AREA TRIM TAB	.60 sq.ft.
TOTAL AREA *(Incl. Balance & Tab)*	13.21 sq.ft.
RUDDER TRAVEL	Right 35 deg. - Left 35 deg.
TRIM TAB TRAVEL	RIGHT 4 deg. - Left 10 deg.

Figure 2—General Airplane Data

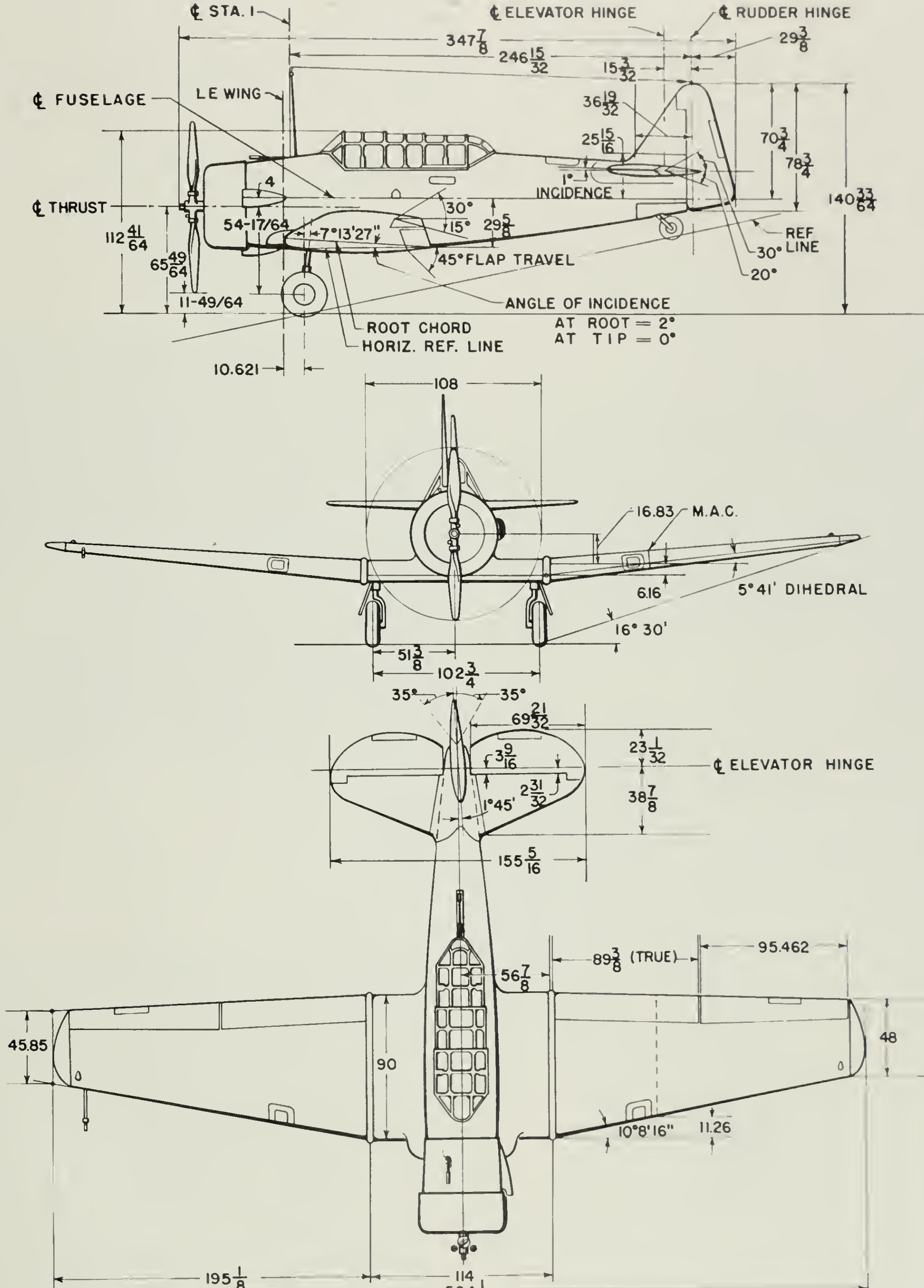


Figure 3—Airplane Three-view Dimensions

No.	Contract Nos. 77 (AT-6A)	Contract Nos. 84 (AT-6B)	Contract Nos. 88 (AT-6C)	Drawing Nos.	Title
	77	84	88	00002	*AIRPLANE ASSEMBLY - GENERAL*
	77	84	88	10001	*INSTALLATION - WINGS*
1.	66			10003	FILLET, WING TO FUSELAGE, FRONT
2.	68			10004	FILLET, WING TO FUSELAGE, INTERMEDIATE REAR
3.	66			10005	FILLET ASSEMBLY, WING TO FUSELAGE, REAR
4.	77	84	88	13000	WING ASSEMBLY, CENTER SECTION COMPLETE
5.	55	84	88	14001	COVERED ASSEMBLY, OUTER WING
6.	55	84	88	14009	ANGLE INSTALLATION, OUTER WING BOLTING
7.	55			14011	TIP ASSEMBLY, WING
8.	78			14112	DOOR ASSEMBLY, OUTER WING LANDING LIGHT
9.	65	84	84	16001	AILERON ASSEMBLY
10.	55			16005	TAB ASSEMBLY, AILERON BOOSTER
11.	55			18001	FLAP ASSEMBLY, OUTER WING
12.	65			18002	FLAP ASSEMBLY, WING CENTER SECTION
	77			20001	*INSTALLATION, EMPENNAGE*
13.	55			20002	FILLET ASSEMBLY, HORIZONTAL STABILIZER LEFT FRONT
14.	55			20003	FILLET ASSEMBLY, VERTICAL STABILIZER TO FUSELAGE
15.	55			20004	FILLET, HORIZONTAL STABILIZER, INTERMEDIATE LOWER
16.	55			20005	FILLET, HORIZONTAL STABILIZER, LEFT REAR
17.	55	–		20008	FILLET, HORIZONTAL STABILIZER, RIGHT REAR
18.	55			20009	FILLET ASSEMBLY, HORIZONTAL STABILIZER, RIGHT FRONT
19.	77	84	84	21001	STABILIZER ASSEMBLY, HORIZONTAL
20.	66			22001	ELEVATOR ASSEMBLY, COVERED
21.	55			22018	TAB ASSEMBLY, ELEVATOR TRIM
22.	55	84	84	23001	STABILIZER ASSEMBLY, VERTICAL
23.	77			24001	RUDDER ASSEMBLY, COVERED
24.	55			24036	TAB ASSEMBLY, RUDDER TRIM
	77	77	88	31001	*FUSELAGE ASSEMBLY COVERED*
25.	77	77	88	31002-17	COWLING ASSEMBLY, ENGINE RING, LOWER
25.	77	77	88	31002-18	COWLING ASSEMBLY, ENGINE RING, RIGHT HAND UPPER
27.	77	77	88	31002-19	COWLING ASSEMBLY, ENGINE RING, LEFT HAND UPPER
28.	77	78	78	31010	COWLING ASSEMBLY, FUSELAGE FLEXIBLE GUN CUTOUT
29.	77	78	78	31013	COWLING ASSEMBLY, FUSELAGE GUNNER'S HOOD
30.	77	77	88	31027	COWLING ASSEMBLY, ENGINE REMOVABLE, CARBURETOR AIR SCOOP
31.	77	77	88	31028	COWLING ASSEMBLY, ENGINE REMOVABLE, UPPER RIGHT
32.	77	77	88	31030	COWLING ASSEMBLY, ENGINE REMOVABLE, BOTTOM
33.	77			31032	FAIRING ASSEMBLY, FUSELAGE FIREWALL UPPER LEFT SIDE
34.	77			31033	COVER ASSEMBLY, FUSELAGE MAIN FUSE BOX ASSEMBLY
35.	77			31034	FAIRING ASSEMBLY, FUSELAGE FIREWALL LOWER LEFT SIDE
36.	77			31035	FAIRING ASSEMBLY, FUSELAGE FIREWALL UPPER RIGHT SIDE
37.	77			31042	FAIRING ASSEMBLY, FUSELAGE FIREWALL LOWER RIGHT SIDE
38.	77			31050	PANEL ASSEMBLY, FUSELAGE LEFT SIDE FAIRING
39.	77	77	88	31051	PANEL ASSEMBLY, FUSELAGE RIGHT SIDE FAIRING
40.	55			31061	COWLING ASSEMBLY, FUSELAGE TAILWHEEL CUTOUT
41.	77			31065	COWLING ASSEMBLY, ENGINE REMOVABLE TOP
42.	77			31066	COWLING ASSEMBLY, ENGINE REMOVABLE LOWER
43.	77			31075	DOOR ASSEMBLY, FUSELAGE FIREWALL TO WINDSHIELD LEFT
44.	77			31076	DOOR ASSEMBLY, FUSELAGE BAGGAGE COMPARTMENT
45.	77			31077	DOOR ASSEMBLY, FUSELAGE FIREWALL TO WINDSHIELD RIGHT
	77	77	88	31101	*FRAME ASSEMBLY, FUSELAGE COMPLETE*
46.	77	78	88	31104	FIREWALL ASSEMBLY, FUSELAGE
47.	77	–		31105	FRAME ASSEMBLY, FUSELAGE FRONT
48.	77			31106	FRAME ASSEMBLY, FUSELAGE REAR SECTION COVERED
49.	55			31238	COVER ASSEMBLY, FUSELAGE REAR SECTION BODY ACCESS
50.	55	78	78	31241	COVER ASSEMBLY, FUSELAGE REAR SECTION FLEXIBLE GUN TROUGH
	77	77	88	31801	*INSTALLATION, COCKPIT ENCLOSURE*
51.	55	55	88	31802	HOOD ASSEMBLY, GUNNER'S
52.	77	77	88	31826	WINDSHIELD ASSEMBLY, COCKPIT ENCLOSURE
53.	55	55	88	31827	PANEL ASSEMBLY, COCKPIT ENCLOSURE, FRONT
54.	49	49	88	31828	PANEL ASSEMBLY, COCKPIT ENCLOSURE, FIXED
55.	55	55	88	31829	PANEL ASSEMBLY, COCKPIT ENCLOSURE, REAR
56.	77			31901	MOUNT ASSEMBLY, ENGINE
	77			33001	*INSTALLATION, LANDING GEAR*
57.	77	78	78	33102	STRUT ASSEMBLY, LANDING GEAR SHOCK
58.	56			33305	FAIRING ASSEMBLY, LANDING GEAR
	55	55	88	34001	*INSTALLATION, TAIL WHEEL*
59.	58	58	88	34002	POST AND HOUSING ASSEMBLY, TAIL WHEEL
	77	78	88	51001	*INSTALLATION, INSTRUMENTS*
60.	77	78	88	51002	PANEL ASSEMBLY, FRONT COCKPIT INSTRUMENT
61.	77	78	88	51003	PANEL ASSEMBLY, REAR COCKPIT INSTRUMENT
	55	55	88	53001	*INSTALLATION, FURNISHINGS*
62.	55	55	88	53010	SEAT ASSEMBLY, REAR COCKPIT COMPLETE
63.	55	55	88	53100	SEAT ASSEMBLY, PILOT'S COMPLETE
	55			73101	*INSTALLATION, PHOTOGRAPHIC EQUIPMENT*
64.	55	55	88	73106	DOOR ASSEMBLY, CAMERA

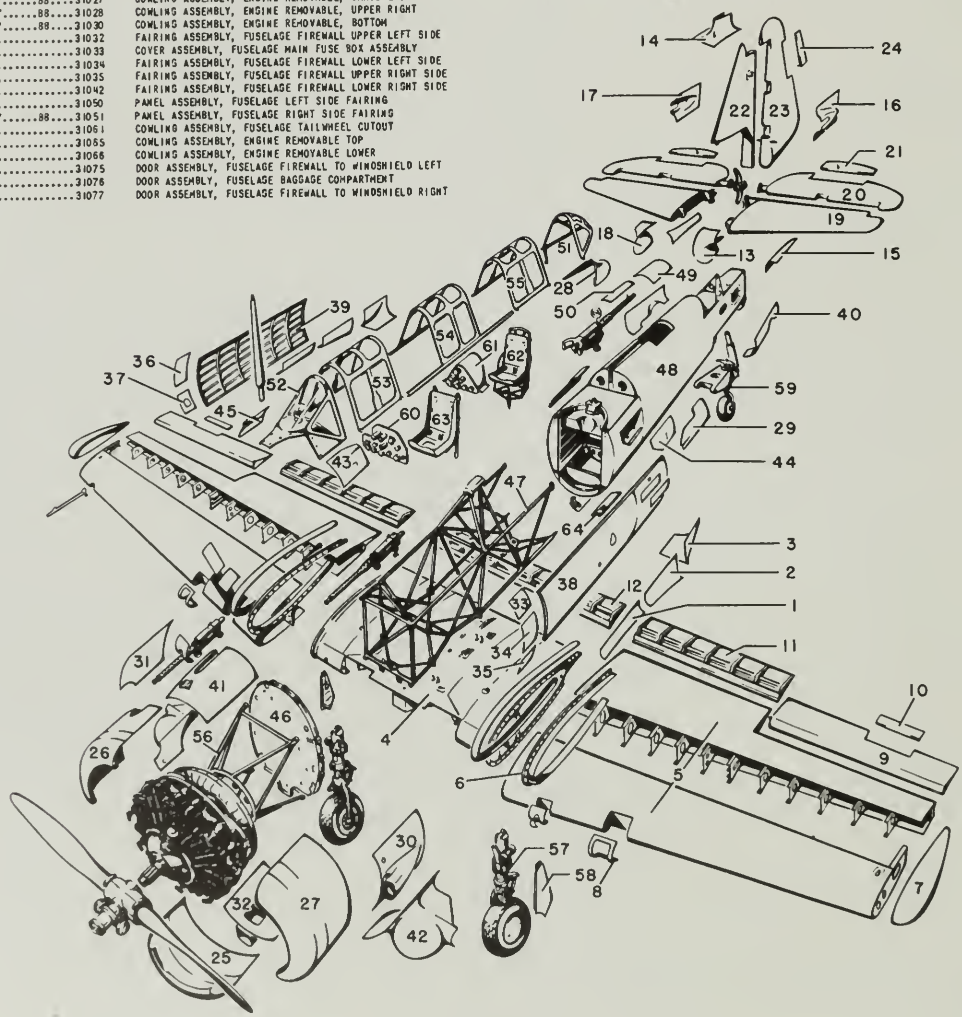


Figure 4—North American Aviation Part Numbers

should be center punched and the actual drilling done with a light power drill. The center-punch mark should be large enough to prevent the drill from slipping out of position, yet must not be made with enough force to dent the surrounding material *(see Figure 5)*. The drilling can be done with a hand drill if no power tool is available, but considerably more time will be taken. Also, as the drill speed is slower when the work is done by hand, there is a tendency to apply more pressure; consequently, a large burr is often formed as the drill pierces the material. All burrs must be removed before riveting by utilizing a metal countersink, or

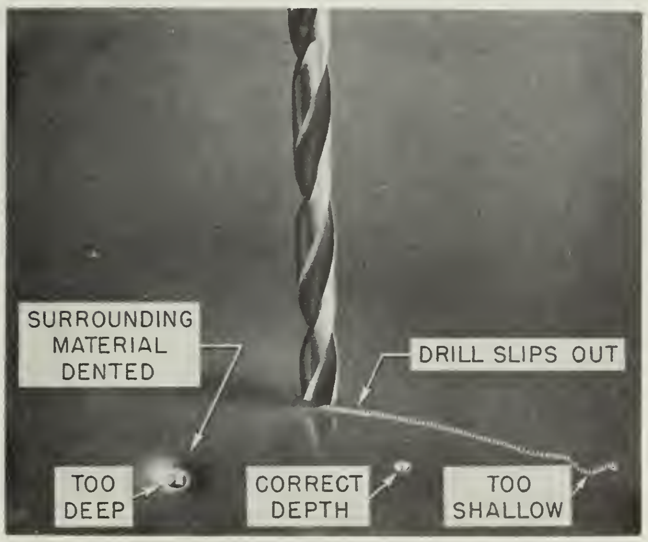


Figure 5—Typical Center-punch Marks

by filing or scraping. In some cases, the use of a right-angled drill is particularly advantageous in drilling through restricted access, small fittings, clips, etc. *(see Figure 6)*.

Figure 6—Using Right-angled Drill

5. STANDARD HOLES FOR RIVETS.

The following chart specifies the size of hole to drill for the application of the various sizes of rivets. See the following paragraph for drill sizes.

RIVET DIAMETER	DRILL SIZE	DECIMAL EQUIV.
3/32	NO. 40	.098
1/8	NO. 30	.1285
5/32	NO. 21	.159
3/16	NO. 11	.191

6. DRILL SIZES.

The following chart specifies the sizes of twist drills from a No. G to a No. 80 with the corresponding decimal equivalents.

SIZE	DEC. EQUIV.	SIZE	DEC. EQUIV.	SIZE	DEC. EQUIV.
G	.2610	5/32	.1562	51	.0670
F	.2570	23	.1540	52	.0635
1/4	.2500	24	.1520	1/16	.0625
D	.2460	25	.1495	53	.0595
C	.2420	26	.1470	54	.0550
B	.2380	27	.1440	55	.0520
15/64	.2344	9.64	.1406	3/64	.0469
A	.2340	28	.1405	56	.0465
1	.2280	29	.1360	57	.0430
2	.2210	30	.1285	58	.0420
7/32	.2187	1/8	.1250	59	.0410
3	.2130	31	.1200	60	.0400
4	.2090	32	.1160	61	.0390
5	.2055	33	.1130	62	.0380
6	.2040	34	.1110	63	.0370
13/64	.2031	35	.1100	64	.0360
7	.2010	7/64	.1094	65	.0350
8	.1990	36	.1065	66	.0330
9	.1960	37	.1040	67	.0320
10	.1935	38	.1015	1/32	.0313
11	.1910	39	.0995	68	.0310
12	.1890	40	.0980	69	.0292
3/16	.1875	41	.0960	70	.0280
13	.1850	3/32	.0937	71	.0260
14	.1820	42	.0935	72	.0250
15	.1800	43	.0890	73	.0240
16	.1770	44	.0860	74	.0225
17	.1730	45	.0820	75	.0210
11/64	.1719	46	.0810	76	.0200
18	.1695	47	.0785	77	.0180
19	.1660	5/64	.0781	78	.0160
20	.1610	48	.0760	1/64	.0156
21	.1590	49	.0730	79	.0145
22	.1570	50	.0700	80	.0135

7. RIVET COUNTERSINKING DIMENSIONS.

The following dimensions should be adhered to when cut countersinking or dimpling sheet material for the application of B1219-AD (or the AN426-AD) 100-degree countersunk rivets. Use the proper degree and diameter countersink and cut just deep enough so the rivet head and the metal will form a flush surface. *(See Figure 7.)*

RIVET SIZE	C'SINK DIAMETER
3/32	11/64
1/8	7/32
5/32	9/32
3/16	11/32

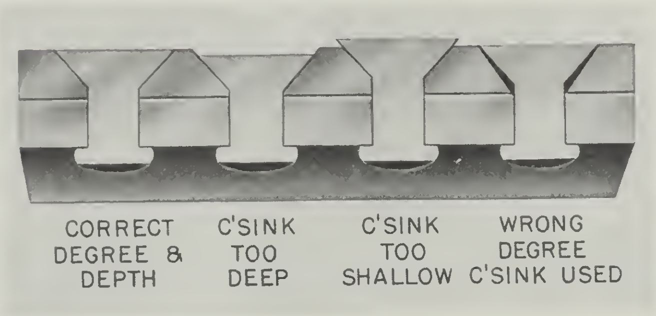


Figure 7—Rivet Countersinking

8. CUT COUNTERSINKING.

When using countersunk rivets in connection with repairs on the forward skin of the fuselage and fixed surfaces, a cut countersinking drill is particularly useful. The cut countersinking attachment consists of a countersinking drill ground to the correct angle and fitted with a small locating tip. It is provided with a carefully made stop and handle. In use, the drill is applied to small pilot holes previously drilled *(See Figure 8)*. For the application of flush rivets, all sheets greater than .032 inch in thickness should be cut countersunk rather than dimpled.

9. DIMPLE COUNTERSINKING.

In connection with the application of countersunk rivets, dimple countersinking is accomplished by the use of a punch and draw die set. The sheet material to be dimpled is placed on the nipple of the punch die and the material is punched into the draw die. The dimple countersinking process should be used only on the thinner gage sheet up to .032 inch in thickness. *(See Figure 9.)*

Figure 8—Cut Countersinking

10. PRERIVETING FASTENERS AND CLAMPS.

Prior to the accomplishment of any riveting, metal sheets must be held securely in position to prevent slipping or creeping during the riveting process. Small "C" clamps are particularly useful; and if both pieces of metal have been

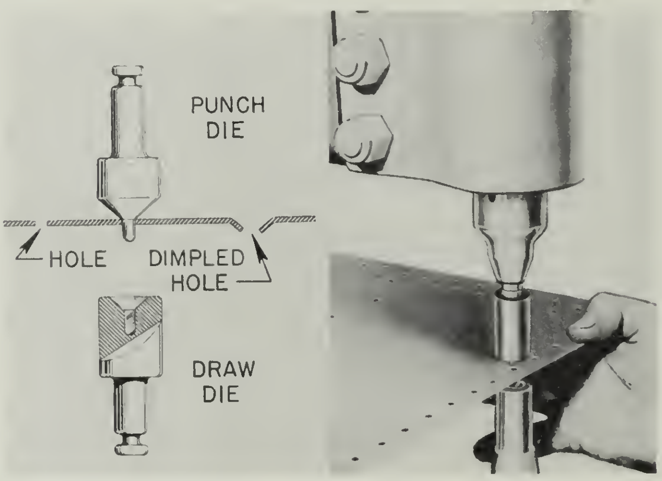


Figure 9—Dimple Countersinking

drilled through, several varieties of skin fasteners are available to line up holes. *(See Figure 10.)* Cleco skin fasteners are commonly

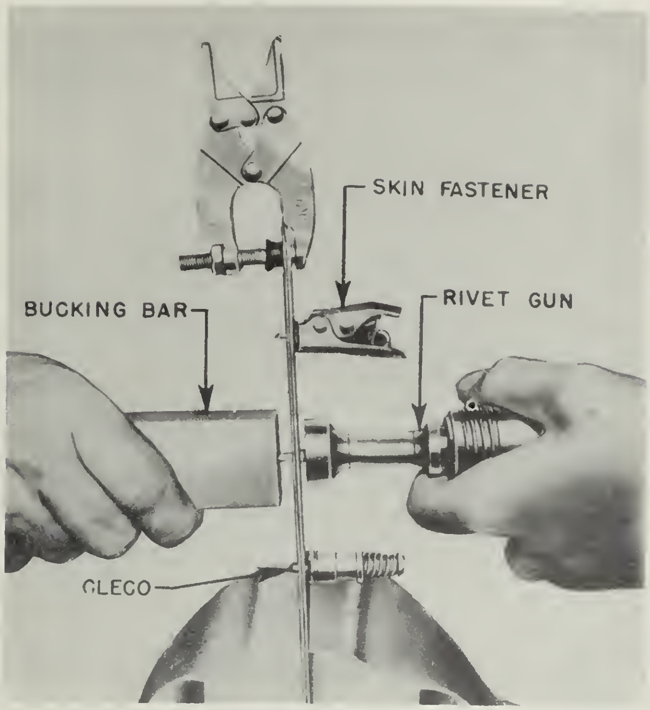


Figure 10—Skin Fasteners

used throughout the aircraft industry in production and repair work. Used in conjunction with a Cleco safety gun *(see Figure 41)*, Clecos hold two pieces of metal firmly together until riveting is accomplished. Inasmuch as Clecos are used only through drilled holes, it is important that the proper size be used. The various sized Clecos are easily identified by their color coding. A Cleco for a 3/32-inch diameter hole is cadmium colored, copper colored for a 1/8-inch hole, black for a 5/32-inch hole, and brass colored for a 3/16-inch hole. *(See Figure 11.)*

11. BUCKING BARS.

Bucking bars may assume any number of given shapes in order to facilitate rivet forming in inaccessible areas. *(See Figure 13.)* The rivet gun pressure applied to the manufactured head of the rivet forms the rivet shank against the bucking bar. In similarity to rivet sets, rivet bucking bars must be kept clean, smooth, and well polished at all times.

12. RIVET SETS.

A rivet set is a punch-like tool with a flat

Figure 11—Cleco Color Coding

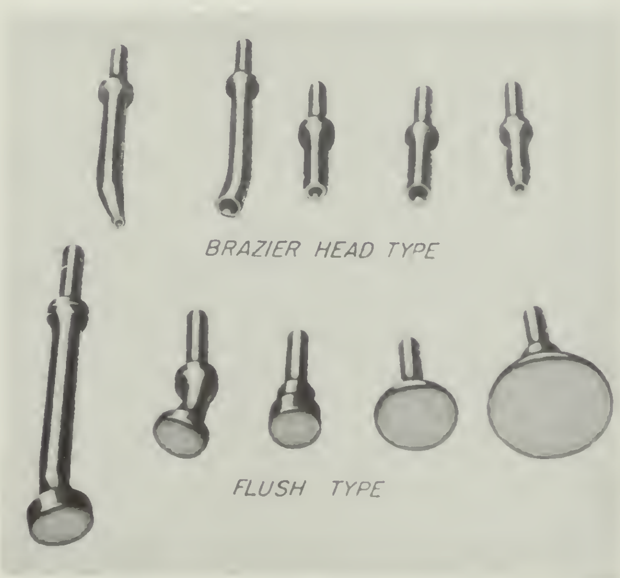


Figure 12—Typical Rivet Sets

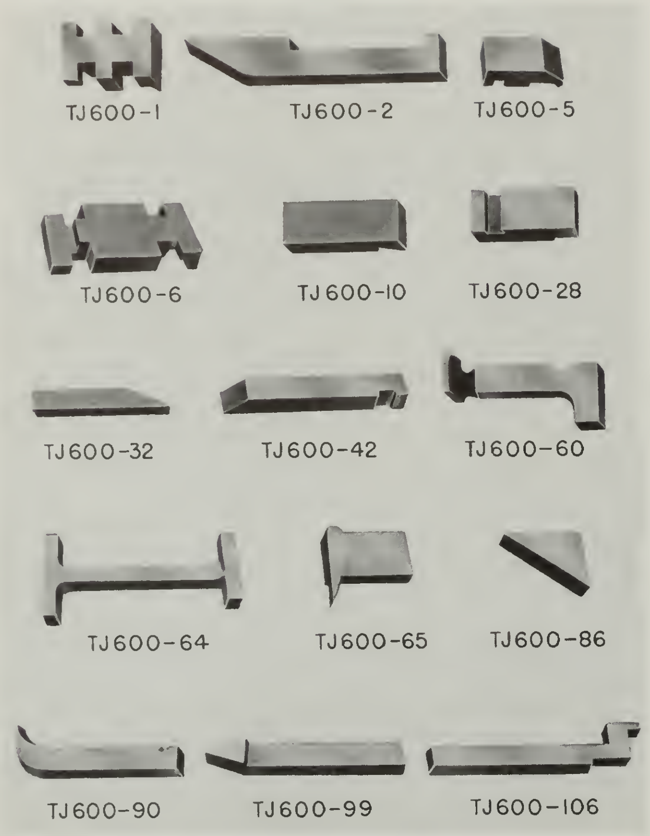


Figure 13—Rivet Bucking Bars

or recessed die for driving flat, countersunk, and brazier head rivets, respectively. They may assume a variety of shapes to facilitate rivet driving in restricted areas *(see Figure 13)*. Rivet sets should be kept smooth and well polished at all times.

13. PNEUMATIC RIVET GUN.

The pneumatic rivet gun used with a rivet set is applied to the manufactured head of the rivet; and the pressure, applied in a series of vibrations, forms the rivet shank against a bucking bar. Different types are available to meet various requirements. *(See Figure 14.)* The period of time during which pressure is applied to form the head must be short enough to prevent the metal from strain hardening near the end of the rivet. The pneumatic equipment may be used either with a flat or recessed rivet set, depending upon the type rivet being driven. Because of the uniformity of driving rivets pneumatically, all riveting should require this equipment. Both the rivet gun and the bucking bar are separate units requiring separate operators. *(See Figure 10.)*

14. HAMMERS.

Hammers sometimes may have to be used to drive rivets, but they should be used only in emergencies where other equipment cannot be obtained. The use of hammers to drive rivets for the repair of primary structure is not recommended. On minor secondary structure, however, ballpeen hammers may sometimes be employed.

15. RIVET SQUEEZER.

A rivet squeezer consists of a large pair of forcers which are used over the edge of the

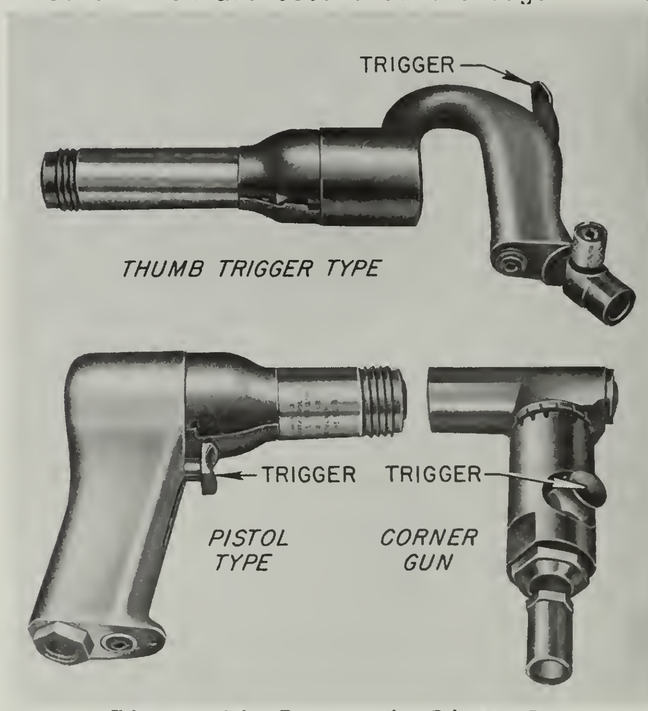


Figure 14—Pneumatic Rivet Guns

material to head the rivet. Pneumatic rivet squeezers may be used with alligator, offset alligator, and C-type jaws. Inasmuch as a rivet squeezer can be used only over the edge of the sheet where conditions permit, its application is necessarily limited. *(See Figure 15.)*

16. REMOVAL OF SOLID RIVETS.

Rivet removal should be accomplished by drilling, not shearing. Use drill one size smaller than diameter of rivet, and place point into depression in rivet head. If power tool is used, locate drill first; then apply power in short bursts. Drill must be applied carefully and small cuts taken in a series of applications. Drill until rivet head twists from shank. *(See Figure 16.)* Drift remainder of rivet free.

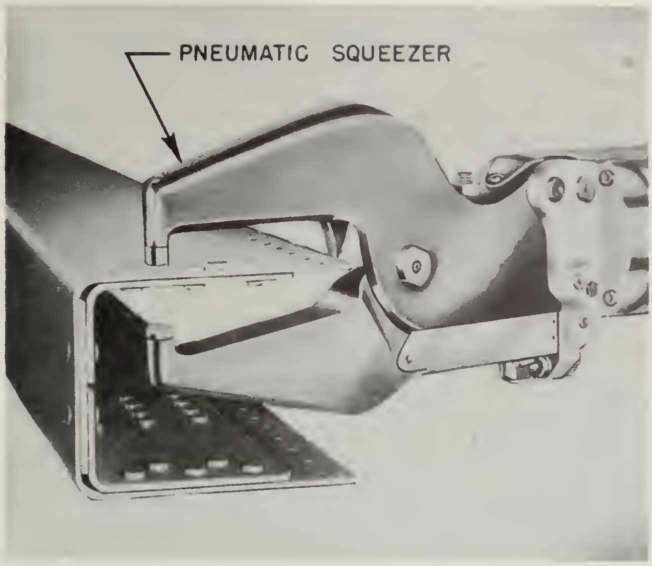


Figure 15—Rivet Squeezer

It is recommended that the drill be equipped with a stop of some soft nonabrasive material to prevent the drill chuck from plunging into and marring the surrounding metal.

17. RIVET AND BOLT EDGE DISTANCES.

The necessity for providing adequate edge distances from the rivet or bolt to the edge of the affected material is of extreme importance in repair. Generally, this distance should not be less than two times the diameter of the rivet or bolt, measured from the centerline of the rivet or bolt hole to the edge of the material involved. *(See Figure 17.)*

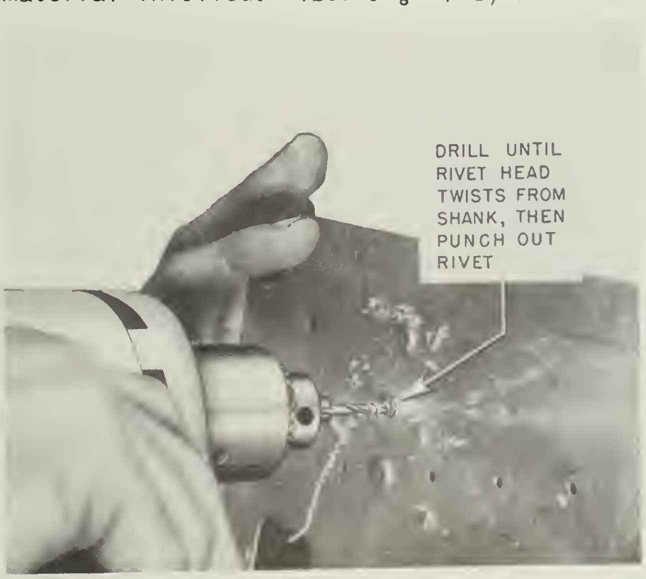


Figure 16—Removal of Solid Rivets

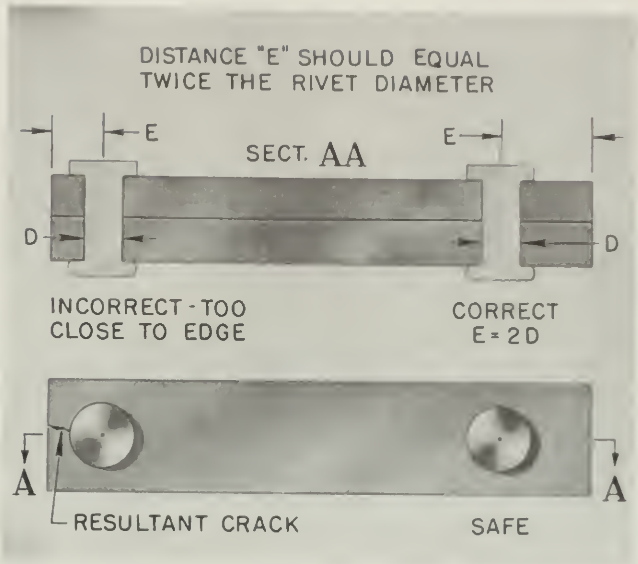


Figure 17—Rivet Edge Distance

18. RIVETING.

Inasmuch as riveting is the major means of securing airplane parts, the quality of workmanship employed in driving rivets must be high. The structural efficiency of the completed product is greatly increased where power-driven tools are available. For this reason, the repair of primary structures should demand pneumatically operated rivet-driving tools. However, for the repair of secondary structures, manually operated rivet-squeezing tools will suffice where conditions permit. The riveting of minor parts sometimes may be accomplished by means of a simple rivet set and 6-ounce ball-peen hammer or the equivalent. Examples of correctly and incorrectly driven rivets are illustrated on the following diagram. *(See Figure 18.)* The following comments are made with reference to this figure.

a. Rivet (A) is driven correctly. The proper length has been determined and the rivet has been driven slowly and compressed evenly. The upset end of an undriven rivet of proper length should measure approximately 1-1/2 times the diameter of the rivet from the surface of the material being riveted.

b. Rivet (B) is driven incorrectly. The head of the rivet has been cut by insecure and/or improperly held driving tool. Rivets damaged in this manner should be replaced. Damage of this nature may be prevented by holding the tool securely and as nearly in alignment with the rivet shank as possible.

c. Rivet (C) is driven incorrectly. The rivet has been driven to excess and/or too much pressure has been applied to the bucking bar. As a result, the head has flattened excessively and cracks are attendant. In the majority of such cases, the cracks extend well into the shank of the rivet. Such conditions are acceptable cause for failure of the rivet upon application of load. Rivets damaged in this manner must be replaced. In the thinner sheet material, the rivet holes become enlarged to the extent that the sheet must be replaced.

d. Rivet (D) is driven incorrectly. This condition is the result of failure to secure the sheets or parts together prior to riveting, thus causing creeping or distortion of the sheet. In this case, the shank of the rivet has swollen into the area produced by separation of the sheets. The rivet must be replaced. Drill the rivet free by means of a drill considerably smaller than its shank diameter; then follow with a drill equal in diameter to the estimated diameter of the swollen portion. Special care should be exercised to prevent such occurrence, as it is most difficult to remove drill chips from between the sheets. Upon reriveting, a smooth surface cannot be obtained. Such conditions greatly lower the value of compressive load resistance of the joint. Various skin clamps can be used in connection with riveting to prevent this occurrence. *(See Figure 10.)*

e. Rivet (E) is driven incorrectly. An unbalanced amount of head material is located about the centerline of the rivet and, as such, the efficiency of the rivet is impaired. This condition is a result of any one or a combination of three things: the rivet bucking bar has not been held securely; it has been forced to one side by pressing too hard without adequate support; or it has been permitted to bounce and slide carelessly over the rivet. This is a most common error and one which should receive much condideration. The rivet should be replaced.

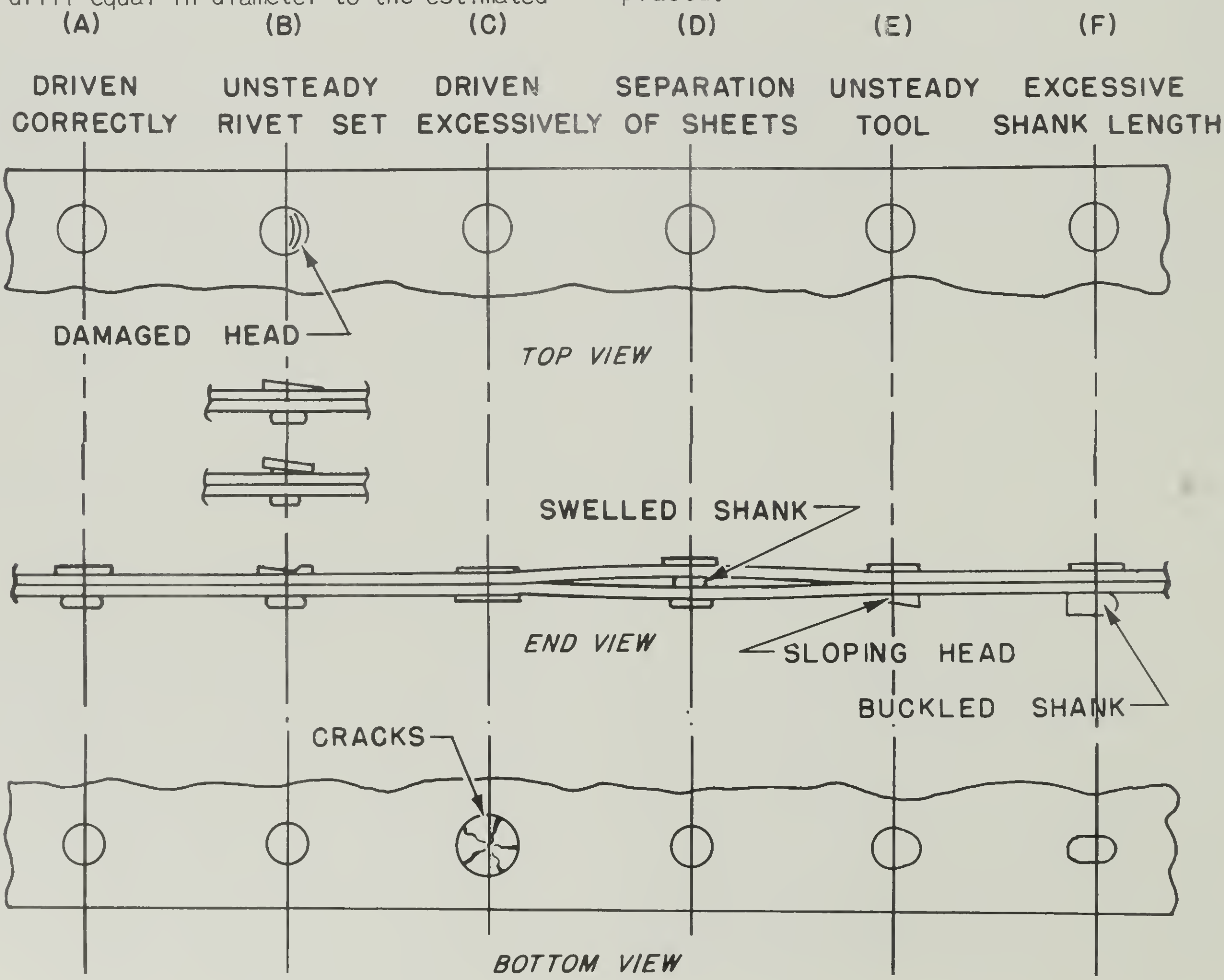


Figure 18—Rivet Driving Practices

f. Rivet (F) is driven incorrectly. The rivet has failed laterally. This condition is a result of improper selection of rivet length, combined with any one or all of the faults of rivet (E). Sufficient care must be exercised in proper selection of rivet length. A rivet too short in length does not permit the required amount of head material to be upset. Too much length results in a buckled shank. Driving a rivet of excessive length also presents possibilities of hole enlargement.

19. FITTING OF BOLTS AND SCREWS.

All bolt holes should be the smallest consistent with design, and all bolts must be suitably locked. Always use the first drill size larger than the nominal diameter of the bolt. When in doubt, it is advisable to drill the hole undersize and ream for a close fit. Bolt holes must not be oversized or elongated. A bolt in an oversized or an elongated hole takes none of the applied shear load until the metal it is holding in shear has slipped enough, and the remaining bolts sheared enough, to allow the bearing surface to come in contact with the bolt. *(See Figure 19.)* This shearing action is a definite failure in these bolts and they are no longer capable of carrying their design loads. In some cases of oversized or elongated holes, it is permissible to drill or ream the hole until large enough to permit the insertion of the next size larger bolt. However, before resorting to this method, obtain permission from an authorized engineering officer. When in doubt, it is advisable to replace the fitting, or fittings, which contains the oversized hole with one in which the holes are the proper size. To make a bolt installation structurally sound, bolt threads must never bear in any part of a fitting. All bolt holes must be normal to the surface involved and must provide complete bearing surface to the head and shank of the bolt. Also, any bolt entering an anchor nut must be free enough to engage the threads in the nut by hand manipulation. Although bolt threads must cut completely through the fiber of self-locking nuts, there should be no more than two bolt threads showing beyond the fiber. To meet this requirement, washers may be used under nuts, and shims may be added under nut plates. The bolts must be sufficiently tight, but too much twist on the wrench must also be avoided. The inch-pounds twist required should be within a certain range for the bolt diameter used. The following chart specifies the twist required. Inch-pounds is obtained by multiplying the pull by the wrench length or by the use of a torque wrench which is calibrated in inch-pounds *(see Figure 20)*.

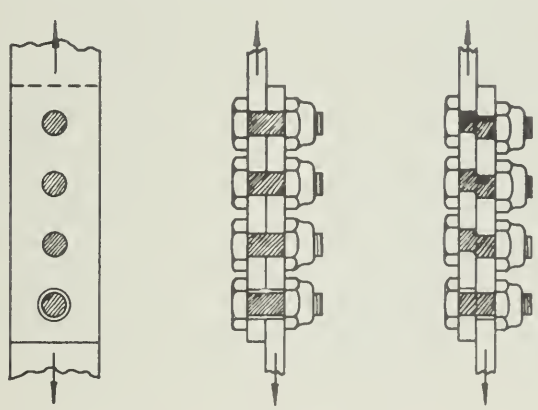


Figure 19—Bolted Joint With Oversized Hole

BOLT SIZE	TWIST (INCH-POUNDS)
AN3 (1032)	20-25
AN4 (1/4)	50-75
AN5 (5/16)	100-140
AN6 (3/8)	160-190
AN7 (7/16)	450-500
AN8 (1/2)	480-690

Figure 20—Bolt Torque Measurement

20. TYPES OF BOLTS, SCREWS, AND NUTS.

With respect to physical properties, bolts, screws, and nuts are of nickel steel and conform to SAE Spec. 2330. Dimension specifications conform to AN and B standards. These specifications, both physical and dimensional, must also be utilized in repair. Generally, only one type of bolt and two types of screws are required in repair. These types are the AN4-type bolt, the B1248, 100-degree countersunk screw, and the B1251 button head screw. *(See Figures 22, 23, and 24.)* The type of nuts used varies with the particular application. General types include the regular self-locking hexagonal, basket anchor, corner anchor, and gang channel *(see Figure 25)*. The self-locking hexagonal nut (elastic stop nut) can be used in oil or gasoline, but it should not be used in the control systems where the bolt would be subject to rotation. For temperatures over 121°C (250°F), the Hy-Temp hexagonal nut should be used, rather than the regular type. It is to be noted that the minumum hexagonal nut height is limited only by the strength requirements of the specification. In some cases, gang channels are particularly useful, but they should not be used in tension applications. The Boot's nuts in the gang channel may be spaced by means of punch marks placed in the channel and they may be readily snapped in or out of the channel. In connection with blind anchoring, the basket anchor nut is also particularly useful, since it eliminates accurate jigging of holes. The corner anchor nut has a tendency to bend thinner gage sheet stock below .040 inch, and the standard two-lug anchor nuts should be used wherever possible *(see Figure 25)*.

21. STANDARD HOLES FOR BOLTS AND SCREWS.

The following chart specifies the sizes of holes to drill for the application of the various sizes of bolts. It is to be noted that bolt holes utilized in repair should be drilled undersize, then reamed for a close fit, wherever possible.

TYPE	DIA.	DRILL		REAM + .0005 − .001
NO. 6	.138	NO. 27	(.144)	
NO. 8	.164	NO. 18	(.1695)	
NO. 10	.190	NO. 14	(.182)	.189
AN4	.250	NO. C	(.242)	.250
AN5	.3125	NO. N	(.302)	.3125

22. ALUMINUM CASTINGS.

Castings of aluminum alloy conform to Aluminum Company of America Specifications Nos. 13, 43, 195-T6, and 356-T6. Castings of 195-T6 material are predominant by virtue of the high general mechanical characteristics produced therein. Castings of a complex, thin-walled character conform to Spec. 356-T6. See Section 12 for the required tensile strength of improvised replacements.

23. ALUMINUM SHEET MARKINGS.

Aluminum sheets may be easily identified by the markings plainly stenciled in rows approximately 3 inches apart on each side of the material. On some sheet stock, only the material composition is conveyed by the markings *(see Figure 21, Detail A)*. On other sheet stock, the composition of the material is given in one line of stencils and the material thickness in thousandths of an inch is given in the other *(see Figure 21, Detail B)*.

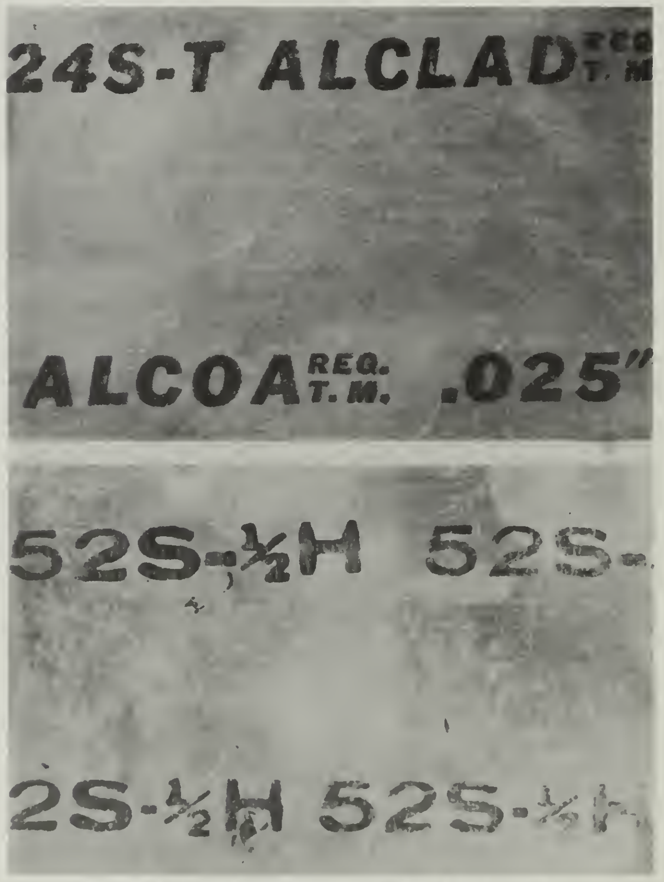
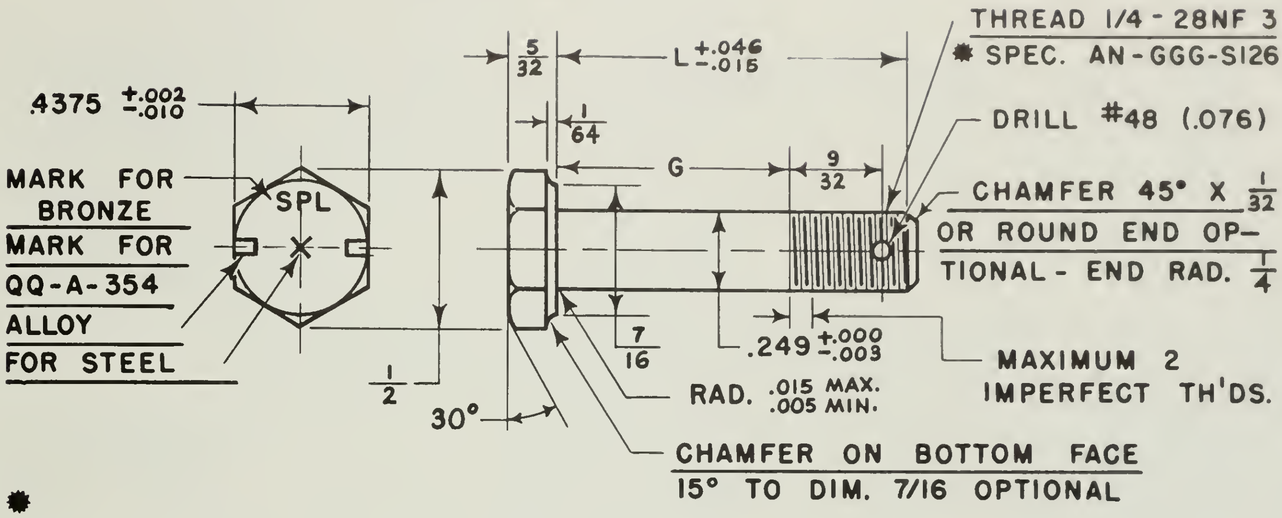


Figure 21—Aluminum Sheet Markings



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DASH NOS.	L'GTH	G	DASH NOS.	L'GTH	G	DASH NOS.	L'GTH	G	DASH NOS.	L'GTH	G
			22	2 1/4	1 13/16	42	4 1/4	3 13/16	62	6 1/4	5 13/16
3	3/8	0	23	2 3/8	1 15/16	43	4 3/8	3 15/16	63	6 3/8	5 15/16
4	1/2	1/16	24	2 1/2	2 1/16	44	4 1/2	4 1/16	64	6 1/2	6 1/16
5	5/8	3/16	25	2 5/8	2 3/16	45	4 5/8	4 3/16	65	6 5/8	6 3/16
6	3/4	5/16	25	2 3/4	2 5/16	45	4 3/4	4 5/16	66	6 3/4	6 5/16
7	7/8	7/16	27	2 7/8	2 7/16	47	4 7/8	4 7/16	67	6 7/8	6 7/16
10	1	9/16	30	3	2 9/16	50	5	4 9/16	70	7	6 9/16
11	1 1/8	11/16	31	3 1/8	2 11/16	51	5 1/8	4 11/16	71	7 1/8	6 11/16
12	1 1/4	13/16	32	3 1/4	2 13/16	52	5 1/4	4 13/16	72	7 1/4	6 13/16
13	1 3/8	15/16	33	3 3/8	2 15/16	53	5 3/8	4 15/16	73	7 3/8	6 15/16
14	1 1/2	1 1/16	34	3 1/2	3 1/16	54	5 1/2	5 1/16	74	7 1/2	7 1/16
15	1 5/8	1 3/16	35	3 5/8	3 3/16	55	5 5/8	5 3/16	75	7 5/8	7 3/16
16	1 3/4	1 5/16	36	3 3/4	3 5/16	56	5 3/4	5 5/16	76	7 3/4	7 5/16
17	1 7/8	1 7/16	37	3 7/8	3 7/16	57	5 7/8	5 7/16	77	7 7/8	7 7/16
20	2	1 9/16	40	4	3 9/16	60	6	5 9/16	80	8	7 9/16
21	2 1/8	1 11/16	41	4 1/8	3 11/16	61	6 1/8	5 11/16			

ENGINEERING INFORMATION

MATERIAL	STAND. PLAIN NUT		STAND. CASTLE NUT COTTER PIN AN380-2-2		TEN. STR. AT ROOT DIA.	YIELD STR. AT ROOT DIA.	SINGLE SHEAR FULL DIA.
STEEL	AN315-4R	MAX.GRIP = G + 1/8	AN310-4	MAX. GRIP = G + 1/8	3980#	3180#	3680#
ALUM.AL. QQ-A-354	AN315D4R		AN310D4		1970#	1270#	1710#

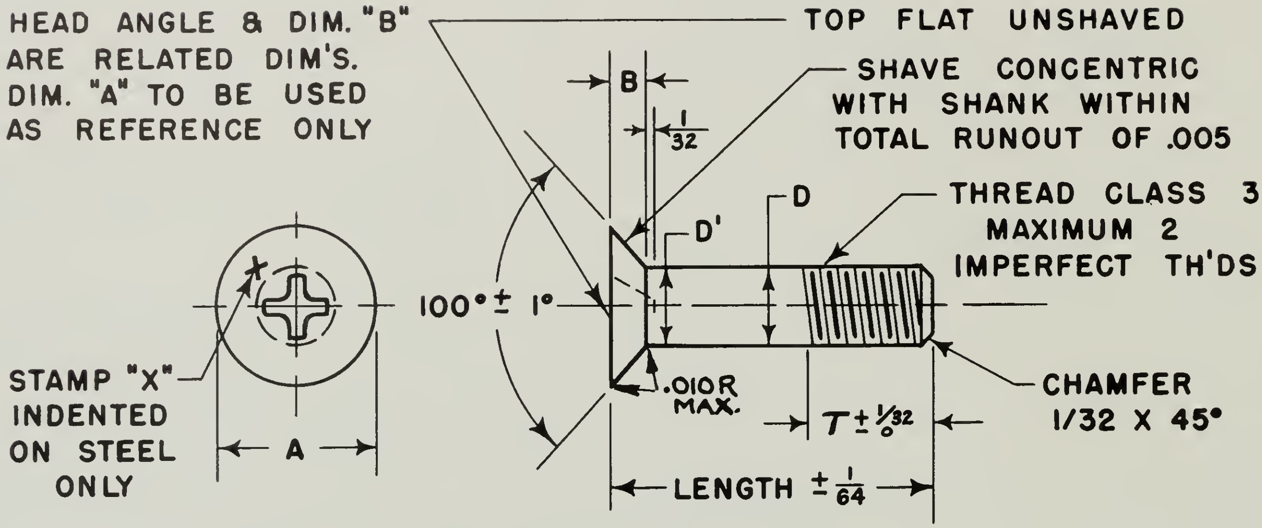
EXAMPLE OF PART NO. - AN4-4 = BOLT - STEEL
AN4DD4 - BOLT - ALUM. ALLOY (QQ-A-354)
ADD "A" TO PART NUMBERS TO DESIGNATE BOLTS WITHOUT COTTER PIN HOLES - AS AN4-4A, AN4D4A, AN4DD4A

SPECIFICATION 29-59

MATERIAL: BRONZE FED. SPEC. QQ-B-746
CODE: SUFFIX "BR" TO PART NO. FOR BRONZE
EXAMPLE: AN4BR-4 = BRONZE BOLT

BRONZE LIMITED TO NONMAGNETIC USE
LIMITS ON DIMENSIONS ± .010 UNLESS OTHERWISE SPECIFIED

Figure 22—Specification AN4 Bolt



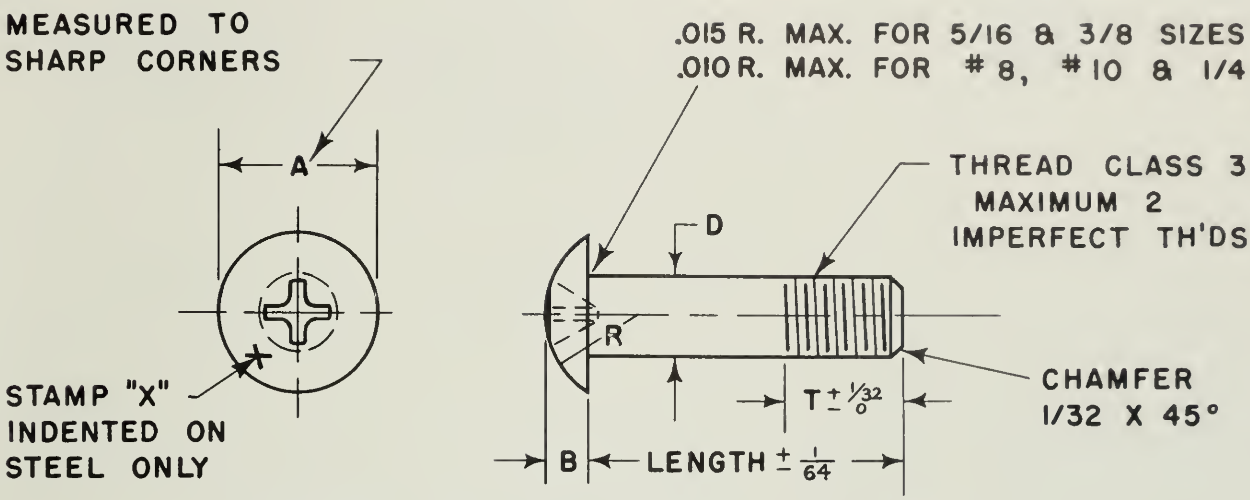
ROLLED OR CUT THREADS OPTIONAL

SIZE	PITCH DIA.	A EXTEND TO EDGE	B ± .033	D + .0000 - .0025	T	PHILLIPS TYPE RECESS DEPTH RECESS	LENGTH RECESS	DRIVER SIZE
8-32	.1437 .1418	5/16	.066	.164	3/8	.098 .088	.166 .156	2
8-36	.1460 .1442	5/16	.066	.164	3/8	.098 .088	.166 .156	2
10-32	.1697 .1678	3/8	.077	.189	3/8	.113 .103	.181 .171	2
1/4-28	.2268 .2246	1/2	.103	.249	7/16	.148 .138	.260 .250	3
5/16-24	.2854 .2830	39/64	.129	.3115	7/16	.183 .173	.332 .322	4
3/8-24	.3479 .3455	3/4	.155	.374	1/2	.208 .198	.357 .347	4
7/16-20	.4050 .4024	7/8	.183	.4365	1/2	.242 .232	.390 .380	4

SUFFIX "BR" TO PART NO. FOR BRONZE
FIRST DASH NO. DENOTES DIA. IN SIXTEENTHS AND THREAD. SECOND DASH NO. DENOTES LENGTH IN SIXTEENTHS.
EXAMPLES: B1248-832-32 SIGNIFIES NO. 8-32, 2 IN. LONG 100° FLAT HEAD SCREW.
B1248-524-46 SIGNIFIES NO. 5116-24, 2-7/8 LONG 100° FLAT HEAD SCREW.

NOTES:
1. DIMENSIONS TO BE MET AFTER PLATING.
2. MINIMUM GRIP B + 1/16.
3. + .001 INCREASE IN D^1 OVER D PERMISSIBLE FOR DIE EXPANSION.
4. THIS SCREW CONFORMS TO NATIONAL AIRCRAFT STANDARD NAS2.
5. CADMIUM PLATE STEEL PER AN-QQ-P-421.
6. HEAT TREAT STEEL 125-140,000 P.S.I.
7. THREAD SPEC. AN-GGG-S-126.
8. DISREGARD "T" WHERE 1/16 MIN. GRIP INTERFERES.
9. MATERIAL - ST'L BAR - ELONG. 17% MIN. AT TENSILE SPECIFIED.
10. BRONZE - HERCULOY 419. REVERE COPPER CO. OR EQUIV., U.T.S. 85,000 MIN.

Figure 23—Specification B1248 Screw



ROLLED OR CUT THREADS OPTIONAL

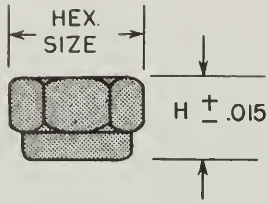
SIZE	PITCH DIA.	A	B	D +.0000 -.0025	R	R	T	PHILLIPS TYPE RECESS DEPTH RECESS	PHILLIPS TYPE RECESS LENGTH RECESS	PHILLIPS TYPE RECESS DRIVER SIZE
8-32	.1437 .1418	.326 .306	.088 .078	.164	.290	.010 .005	3/8	.100 .090	.155 .145	2
8-36	.1460 .1442	.326 .306	.088 .078	.164	.290	.010 .005	3/8	.100 .090	.155 .145	2
10-32	.1697 .1678	.385 .365	.102 .090	.189	.320	.015 .007	3/8	.115 .105	.170 .160	2
1/4-28	.2268 .2246	.510 .490	.127 .115	.249	.340	.020 .010	7/16	.155 .145	.245 .235	3
5/16-24	.2854 .2830	.573 .553	.163 .149	.3115	.390	.025 .012	7/16	.193 .183	.310 .300	4
3/8-24	.3479 .3455	.698 .678	.194 .180	.374	.484	.030 .015	1/2	.234 .224	.349 .339	4

CODE:
SUFFIX "BR" TO PART NUMBER FOR BRONZE
FIRST DASH NUMBER DENOTES SIZE AND THREAD.
SECOND DASH NUMBER DENOTES LENGTH IN SIXTEENTHS.

EXAMPLES:
B1251-832-32 SIGNIFIES 8-32 X 2 IN. LONG RECESSED BUTTON HEAD SCREW.
B1251-524-46 SIGNIFIES 5/16-24 X 2-7/8 IN. LONG RECESSED BUTTON HEAD SCREW.

NOTES:
1. DIMENSIONS TO BE MET AFTER PLATING.
2. MINIMUM GRIP 1/16.
3. THIS SCREW CONFORMS TO NATIONAL AIRCRAFT STANDARD NAS3.
4. THIS SCREW FORMERLY B1225 TYPE W. (EXCEPT AS NOTED.)
5. CADMIUM PLATE STEEL PER AN-QQ-P-421.
6. THREAD SPEC. AN-GGG-5-126.
7. HEAT TREAT STEEL 125-140,000 P.S.I.
8. DISREGARD "T" WHERE 1/16 MIN. GRIP INTERFERES.
9. MATERIAL ST'L. BAR ELONG. 17% MIN. AT TENSILE SPECIFIED.
10. BRONZE - HERCULOY 419, REVERE COPPER AND BRASS CO. OR EQUIV. U.T.S. 85,000 MIN.

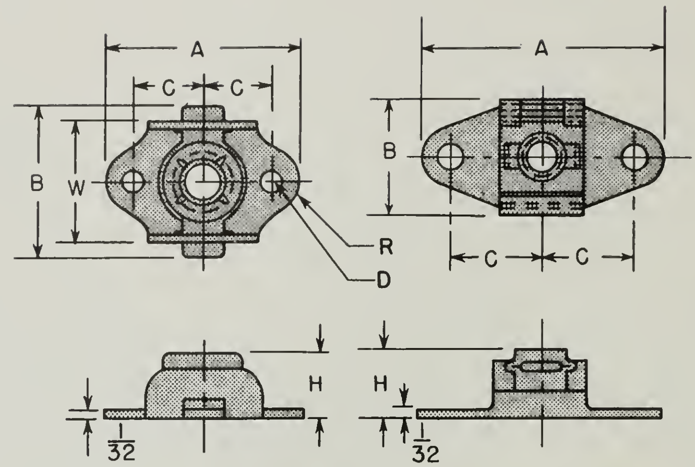
Figure 24—Specification B1251 Screw



DIMENSIONS

SIZE & THREAD	HEX. SIZE	H ±.015	STEEL CAD PLATING
4-40	1/4	9/64	AC365-440
6-32	5/16	11/64	AC365-632
8-32	3/8	15/64	AC365-832
10-32	3/8	15/64	AC365-1032
1/4-28	7/16	5/16	AC365-428
5/16-24	1/2	11/32	AC365-524
3/8-24	9/16	29/64	AC365-624
7/16-20	5/8	29/64	AC365-720

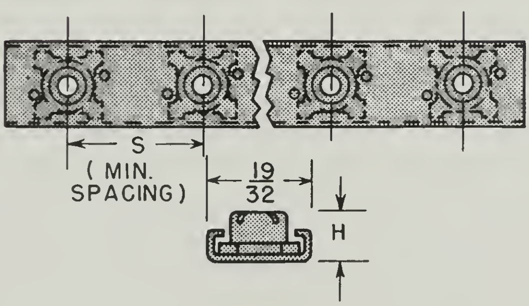
REGULAR HEXAGONAL



DIMENSIONS — PART NUMBERS

SIZE & THREAD	A	B	C	D	H	R	W	1112 STEEL NUTS AND BASKETS
6/32	31/32	43/64	11/32	.098	5/16	9/64	35/64	22A21-62
8/32	31/32	43/64	11/32	.098	21/64	9/64	35/64	22A21-82
10/32	31/32	43/64	11/32	.098	11/32	9/64	35/64	22A21-02
1/4-28	1-9/32	43/64	1/2	.098	13/32	9/64	39/64	22A21-048
5/16-24	1-9/32	51/64	1/2	.103	25/64	9/64	43/64	22A21-054

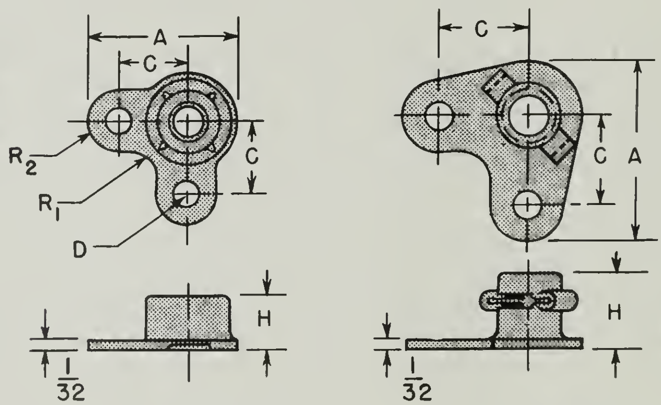
BASKET ANCHOR



DIMENSIONS

SIZE & THREAD	S	H	STEEL CHANNEL & NUTS	AL. ALLOY CHANNEL & STEEL NUTS	STEEL NUTS ONLY	STEEL CHANNEL ONLY
8/32	5/8	.297	22G1-82-5	68G1-82-5	22G1-82	22GCH-82
10-32	5/8	.313	22G1-02-5	68G1-02-5	22G1-02	22GCH-02
1/4-28	3/4	.375	22G1-048-6		22G1-048	22GCH-048
5/16-24	1	.375	22G1-054-8		22G1-054	22GCH-054
3/8-24	1	.4375	22G1-064-8		22G1-064	22GCH-064

GANG CHANNEL



DIMENSIONS — PART NUMBERS

SIZE & THREAD	A	C	H	R_1	R_2	D	STEEL CAD PLATED
6-32	11/16	11/32	13/64	13/64	9/64	.098	22A5-62
8-32	11/16	11/32	9/32	13/64	9/64	.098	22A5-82
10-32	11/16	11/32	5/16	13/64	9/64	.098	22A5-02
1/4-28	7/8	1/2	23/64	15/64	9/64	.098	22A5-048
5/16-24	29/32	1/2	23/64	17/64	9/64	.130	22A5-054
3/8-24	61/64	1/2	15/32	5/16	9/64	.130	22A5-064

CORNER ANCHOR

Figure 25—Types of Nuts

24. ALUMINUM WROUGHT STOCK - GENERAL.

The symbol S is used after the formula number of wrought alloys to distinguish them from the cast alloys. In the case of heat-treatable aluminum, the symbol O following the S indicates an annealed material, but the symbol T following the S indicates a heat-treated, hardened material. Alloy 24ST in the form of sheet and extruded shapes is used almost entirely in the construction of the aircraft; but because of better forming qualities, parts are often fabricated of 24SO and subsequently heat treated. Alloys 14ST and 17ST are used for forgings only. Material of 14ST, possessing high mechanical properties, is most prevalent in forgings. Material of 17ST finds use where an increase in corrosion resistance is required, with but a slight sacrifice in strength. See Section 12 for the required strength of forgings. The symbol W applies only to alloys requiring artificial aging and is used principally with the alloy 53SW to signify a heat-treated, quenched, but not completely age-hardened material. It is to be noted that the use of the symbol T to designate a hard alloy is applicable only to the heat-treatable alloys, principally 14ST, 17ST, and 24ST. However, of the wrought alloys, there are a few which are nonheat-treatable, principally 2S, 3S, and 52S. These nonheat-treatable alloys are also nonstructural; but because of their malleability, they find particular use in fairing and filleting. Cold working hardens these alloys to a certain extent and this condition is noted not by the symbol T, but by the symbol H. Intermediate conditions of hardness are noted as 1/4H, 1/2H, etc. The following chart gives general particulars pertinent to wrought alloy uses.

ALLOY	FORM	USE
14ST 17ST	FORGINGS	FITTINGS
24ST	SHEET, ALCLAD	SKIN, SPARS, LONGERONS, RIBS, FRAMES, FLOORS, FUSELAGE STRINGERS
24ST	EXTRUDED SHAPES	SPAR CAPS, WING STRINGERS, STIFFENERS
53ST	EXTRUDED SHAPES	ENCLOSURE & WINDOW FRAMES ONLY
53SW	TUBE	PLUMBING, LINES
2S	SHEET	FAIRING, FILLETING
3S	SHEET	FAIRING, FILLETING
52S	SHEET & TUBE	FAIRING, PLUMBING, LINES

25. ALUMINUM SHEET, 2SO.

This material is the wrought form of commercially pure aluminum. It is relatively low in strength, but possesses high corrosion resistance and is extremely easily formed and worked.

26. ALUMINUM SHEET, 2S-1/2H.

This material possesses the same chemical content as 2SO. However, it is hardened to a point halfway between the soft and hard tempers by means of cold working. The material is less workable and presents some increase in stiffness over 2SO. The main usage of 2S-1/2H material is in electrical junction boxes. These boxes are drawn and require some structural stiffness. The electrical conductivity of the material produces an excellent value of radio shielding.

27. ALUMINUM SHEET, 3SO.

This material possesses much the same qualities of corrosion resistance as the 2S alloys. It tends to produce somewhat better structural possibilities than the 2S alloys and has forming values only slightly inferior. It finds particular adaptability in fillets where forming more severe than is possible by harder materials is necessary.

28. ALUMINUM SHEET, 3S-1/2H.

This alloy is identical to the 3SO material in chemical content and, in similarity to 2S-1/2H alloy, is hardened by cold working. It is employed for filleting where forming of severe nature is a primary requisite.

29. ALUMINUM SHEET AND TUBE, 52SO.

This material possesses tensile strength qualities approximately twice that of 2SO alloy. The forming properties are midway between those of 2SO and 3SO materials. The material is used most generally for fillets and fairing where an important function is workability. Tube stock of 52SO material is used in most of the systems of the airplane. This type of material is highly resistant to fatigue brought about by vibration, and does not require annealing after prolonged service.

30. ALUMINUM SHEET, 52S-1/2H.

This alloy possesses the same chemical and mechanical properties as those of 52SO; but in similarity to the 2S-1/2H and 3S-1/2H alloys, it is hardened to an intermediate temper between the soft and hard by cold working. The material finds general use in fillets and fairing where intermediate values of strength, hardness, and ductility are required.

31. ANNEALING ALUMINUM, 2S, 3S, AND 52S.

The nonheat-treatable alloys, 2S, 3S, and 52S, are sufficiently malleable for cold working in their original state. If these materials have already been strain hardened by cold working, annealing is instantaneous upon reaching the proper temperature, and the material should be promptly removed from the furnace and allowed to cool at normal room temperature. Hardness is accomplished only by cold working. The following chart specifies annealing temperature particulars.

ALLOY	ANNEALING TEMPERATURE	
	°C	°F
2S	329-371	625-700
3S	385-413	725-775
52S	329-371	625-700

32. ALUMINUM EXTRUSIONS, 24SO.

This material has values of corrosion resistance considerably inferior to 2S or 3S, but it possesses excellent working and forming qualities. The material is heat-treatable, but it does not work harden. Heat treatment brings its strength and hardness to a level fully equal to that of 24ST material. All 24SO material must be heat treated after forming.

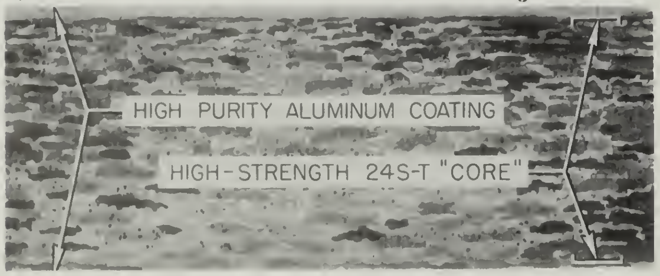


Figure 26—Enlarged Cross Section of Alclad Sheet

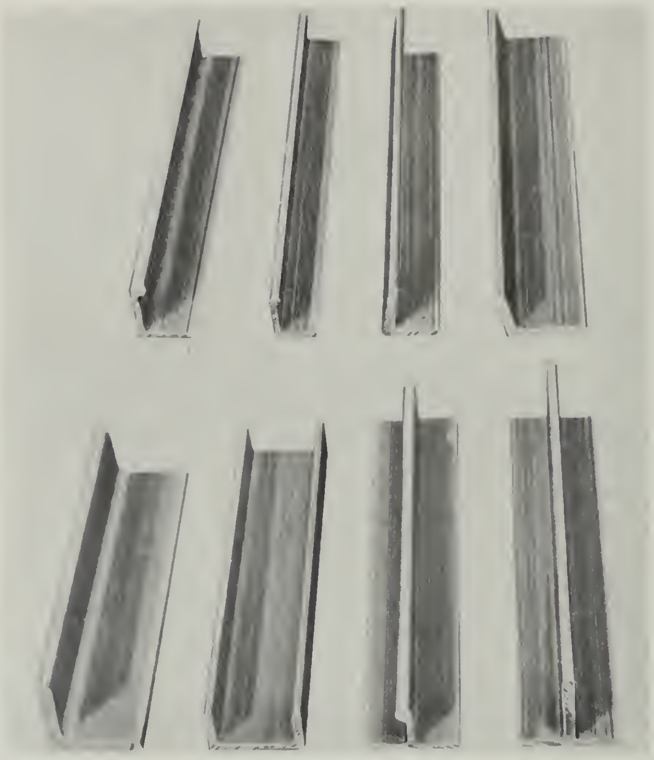


Figure 27—Typical Extruded Sections

33. ALUMINUM EXTRUSIONS, 24ST.

This material is the hardened form of 24SO aluminum alloy. Extruded shapes of 24ST are extensively employed as stringers, longerons, and spar caps. *(See Figure 27.)* The 24ST aluminum alloy composition is the strongest and hardest of all the alloys used, but the corrosion resistance of the material is considerably inferior to 24S alclad sheet. To protect the material against corrosion, zinc chromate primer must be applied to all surfaces.

34. ALUMINUM SHEET, 24SO ALCLAD.

This material is identical to 24SO aluminum alloy; but in order to provide greater protection against corrosion, a thin layer of pure aluminum 2S is bonded to each side of the sheet. This pure aluminum coating has a very high resistance to direct contact with corrosive agents and also, by electrolytic action, it serves to protect the cut edges of the sheet. The material is slightly inferior in strength to 24SO aluminum alloy, but it possesses excellent forming qualities. Parts requiring severe working and bending should be formed of 24SO alclad and then heat treated to the full hardness of 24ST.

35. ALUMINUM SHEET, 24ST ALCLAD.

The heat-treated form of 24SO alclad is known as 24ST. This material is the most generally used of all the aluminum alloys and is utilized for ribs, frames, spars, longerons, stringers, skin, and formers. Although slightly inferior to 24ST aluminum alloy in strength, 24ST alclad possesses far greater corrosion resistance. On each face of the high-strength 24ST core is an integrally bonded layer of relatively high purity aluminum known as the coating. *(See Figure 26.)* This coating protects the core in two ways. It protects the core electrolytically and covers the core to prevent contact with corrosive agents. The material should never be fully annealed.

36. HEAT TREATING ALUMINUM, 24SO.

The heat treating of 24SO aluminum or 24SO alclad should be accomplished in either an air furnace or salt bath furnace. Methods of treatment in which the flame may come in direct contact with the material are not recommended. The heat-treating process consists essentially of: (1) heating the material to a prescribed temperature; (2) holding the material at this temperature for a specified length of time; (3) promptly quenching the material; and (4) allowing the material to age harden at room temperature for a specified period. The actual heat-treating temperature of 24SO aluminum and 24SO alclad must remain constant within a range of 491-499°C (915-930°F) for the time specified on the following charts. However, periods specified should be timed after the charge has reached uniform heat-treating temperatures throughout the furnace. Also, when heat treating parts of varied thickness simultaneously, use the thickest sheet as a timing criterion. For example, when heat treating a sheet of .040-in. and a sheet of .064-in. use a minimum heat-treating time of 30 minutes and maximum heat-treating time of 40 minutes.

AIR FURNACE METHOD

MATERIAL THICKNESS (INCHES)	MINUMUM TIME (MINUTES)	MAXIMUM TIME (MINUTES)
UP TO .020	5	15
.021 - .032	10	20
,033 - .063	20	30
.064 - .125	30	40
.126 - .250	50	70
.251 - .500	90	110

SALT BATH METHOD

MATERIAL THICKNESS (INCHES)	MINIMUM TIME (MINUTES)	MAXIMUM TIME (MINUTES)
UP TO .020	5	15
.021 - .032	10	20
.033 - .063	15	25
.064 - .250	20	30
.251 - .500	30	40

Figure 28—Quenching Heat-treated Part

37. QUENCHING.

It is essential that heat-treated material be quenched within 8 seconds after removal from the furnace. A greater delay in the transfer from the furnace to the quenching bath materially affects the value of corrosion resistance. The parts may be directly immersed in water or held in a dense cold water spray; but in any case, the temperature of the water should not exceed 38°C (100°F) during this operation. *(See Figure 28.)* If a salt bath furnace is employed, the parts must be thoroughly washed in mildly warm water after quenching to remove all traces of salt. A certain amount of warpage (due to the quenching) should not cause rejection if the part can be properly straightened during assembly or within one hour after quenching. In fact, if any rework is necessary, it is advisable to accomplish it always within one hour after quenching before the aging has progressed too far. During this period, the metal may be worked with ease and without danger of cracking. Aging may be retarded by placing the material in a cold atmosphere.

38. AGE HARDENING.

After quenching, the strength of the 24S aluminum or 24S alclad gradually increases at room temperature until, after approximately 24 hours aging, the material reaches a fully hardened condition. When this occurs, the material is ready for installation.

39. ANNEALING ALUMINUM, 24ST.

When annealed 24S or 24S alclad materials are required in the course of repair, they should be obtained in the annealed condition

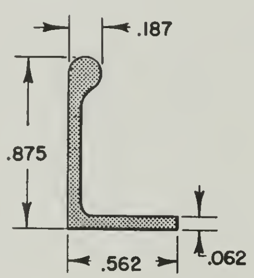
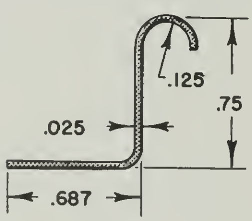
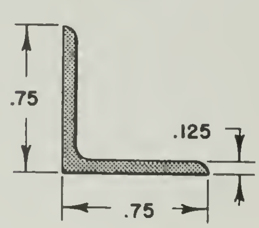
NAA STANDARD	*ALCOA DIE NO.*	*PRINCIPAL DIMENSIONS*
C366T	L24910	
C373LT (ROLLED ALCLAD)	NONE	
K77A	K77A	

Figure 29—Extrusion Equivalent Chart

from the metal manufacturer. However, if 24SO material is not available, it is sometimes permissible to anneal 24ST aluminum alloy fully and to anneal 24ST alclad partially. However, this is not recommended where the material is to be used in location of high stress. Of the temperature range specified for annealing 24ST aluminum alloy, the lower temperature (343°C, 650°F) will anneal the material sufficiently to permit removal of cold-working strains as required for all ordinary types of repair fabrication. The upper temperature (427°C, 800°F) specified below for 24ST aluminum alloy will permit maximum annealing; however, the material must be held at this temperature for two hours, then cooled to 260°C (500°F) at the rate of 10°C (50°F) per hour. After the temperature of aluminum alloy has reached 260°C (500°F), the cooling rate in air is not critical. It is to be noted that the temperature range specified below for 24ST alclad will only partially anneal the material. Full annealing of 24ST alclad is not permitted, because of the rapid diffusion of the base metal through the aluminum coating occurring as a result of slow cooling from high temperatures. The following chart specifies annealing temperature particulars applicable to air furnace and salt bath furnace.

ALLOY	ANNEALING TEMPERATURE	
	°C	°F
24ST AL. ALLOY	343-427	650-800
24ST ALCLAD	338-349	640-660

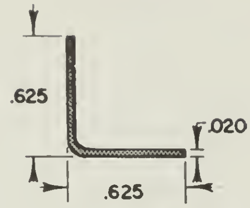
40. BENDING 24S ALCLAD.

Exercise caution in handling alclad materials to avoid scratching through the pure aluminum coating. Besides destroying the protective aluminum coating, scratches in alclad material are potential cracks which may develop under the application of stress. Where the material must be flanged, a power or hand brake will be found particularly useful. Alclad sheet may be formed and bent as required, but the allowable bend radius for the thickness of sheet, as specified herein, should not be exceeded. The following chart specifies particulars pertinent to the bending of alclad.

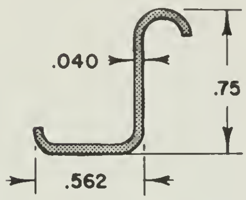
ALCLAD	ALLOWABLE BEND RADIUS
24ST	4 TIMES THE THICKNESS OF THE SHEET OR GREATER.
24SO	2 TIMES THE THICKNESS OF THE SHEET OR GREATER.

NAA STANDARD — ALCOA DIE NO. — PRINCIPAL DIMENSIONS

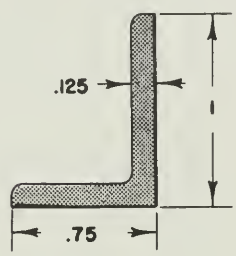
C107LT-20 (ROLLED ALCLAD) — NONE



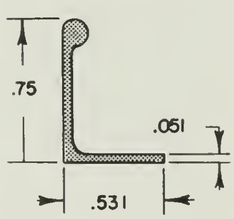
C123LT (ROLLED ALCLAD) — NONE



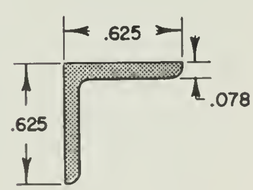
C148T — K734-JJ



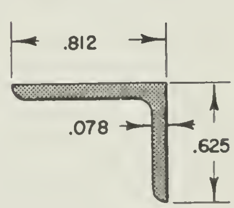
C180T — K14280



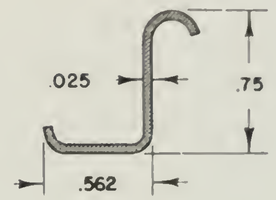
C203T — K14653



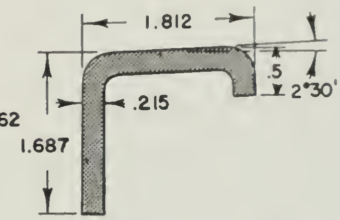
C204T — K14654



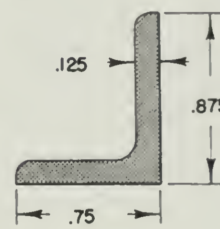
C234LT (ROLLED ALCLAD) — NONE



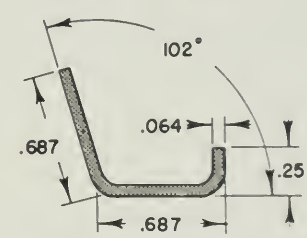
C245T ONE-HALF SIZE — K16862



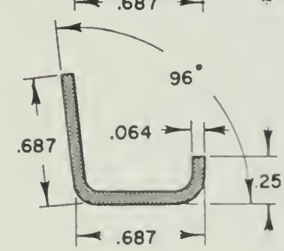
C250T — K16869



C265T — L23557



C266T — L23558



C274T — L23966

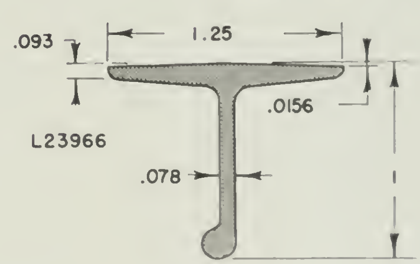


Figure 29A—Extrusion Equivalent Chart (contd.)

41. EXTRUSION EQUIVALENT DIE NUMBERS.

In order to expedite the procurement of the extrusions required in the repair of this series airplane, North American Aviation standard part numbers and equivalent die dimensions and numbers of the Aluminum Company of America (Alcoa) are incorporated in this Manual *(see Figure 29)*. It is to be noted that all extruded members are made of 24ST aluminum alloy material.

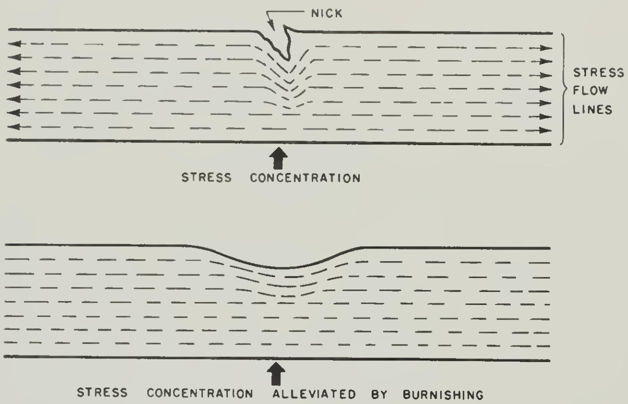


Figure 30—Stress Concentrations

42. SCRATCHES, DENTS, AND CRACKS.

Many parts become scratched and dented in the course of service, thus producing stress concentrations which may cause failure of the part. Sharp nicks in the smooth surface are especially dangerous, and care must be taken at all times to smooth out such damage. When this is done high concentrations of stress disappear. *(See Figure 30.)* The importance of smoothing out surface irregularities cannot be overemphasized because the working stresses are usually so high that any defect of this kind may produce stresses above the strength of the part. Care should be taken to ensure that no cracks remain undetected. Cracks in the skin generally occur at points of cold working and when the skin is drilled or dimpled for flush-type rivets or bolts. To arrest the growth of the crack, drill a No. 40 hole at each end of the crack and clean out any rough material around the crack.

43. SUPPORT OF STRUCTURE DURING REPAIR.

It is essential that the structure to be repaired is adequately supported against distortions when any member, or part of a member, is removed for assembly. For instance, it is very important that the wing be adequately supported when removing a panel of wing skin.

44. EXTENT OF DAMAGE.

When estimating the extent of damage, the surrounding structure must be carefully examined to ensure that no secondary damage remains unnoticed. Secondary damage may be produced in some structure remote from the location of the primary damage by the transmission of the damaging concentrated load over the normal course toward the basic reacting structure. Damage of this nature usually occurs where the most abrupt change in load travel is experienced. If this damage remains undetected, loads applied in the normal course of operation may cause failure of the part. It is to be noted that all loads travel toward the structure comprising the wing attachment to the fuselage. Carefully investigate all structure across which the loads must travel to reach these points. Particularly examine joints for security of rivets, bolts, bent plates, etc.

45. CORROSION PROTECTION.

When repairing parts, adequate measures must be taken to prevent the subsequent occurrence of corrosion. See Section II for particulars.

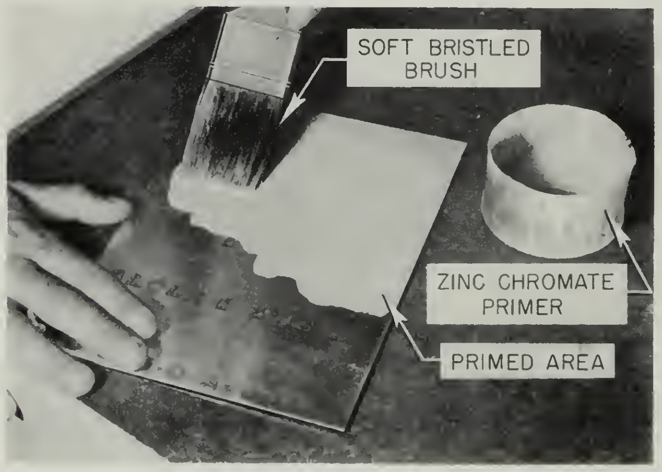


Figure 31—Priming Aluminum Sheet

One coat of zinc chromate primer should be applied to all surfaces of extruded repair members and to the overlapping surfaces of all 24ST alclad sheet repair members. Always stir zinc chromate primer thoroughly before use and apply an even coat with a soft bristled brush. *(See Figure 31.)*

46. TOOLS - GENERAL.

Wherever possible, power tools should be used for repairs; but where facilities are limited, hand tools sometimes may be employed. Riveting tools are needed most frequently and these have been described in preceding paragraphs. Occasions arise when repairs are impractical and a replacement part must be used in order to restore the component to normal operating condition. Other occasions arise when an assembly will be made more accessible and hence more conveniently repaired if removed from the airplane. For all such occasions, a standard tool kit containing an assortment of tools will be necessary *(see Figure 32)*.

Figure 32—Standard Tool Kit

47. METAL SHRINKER OR STRETCHER.

Replacement sections for curved fairing, fillets, and cowling may be readily fabricated by utilizing a metal shrinker and/or stretcher. The foot-lever-operated machine has an upper and a lower set of movable jaws, which move vertically and horizontally, shrinking or stretching the metal. These two operations are accomplished by the use of two interchangeable sets of jaws and side plates. To operate this machine, place the sheet between the jaws, and move the sheet along in successive uniform movements while operating the foot lever. To increase the sheet curvature and to decrease circumference, use shrinker jaws. To decrease curvature and to increase circumference, use stretcher jaws. *(See Figure 33.)*

Figure 33—Metal Shrinker and Stretcher

48. HAND ROLLER.

The hand roller used for rolling and beading sheet stock consists of a handle and a set of gears, which in turn are attached through shafts to rollers, interchangeable as to size, shape, and depth of grooves. Several effects can be obtained with this tool, the most common of which is the beading of sheet stock. A guide is provided for lining up the metal and a crank is employed to raise or lower the upper roller for a deep or shallow bead as required. To operate, adjust the depth to suit, place edge of material on face of guide, push sheet snugly between rollers, and turn actuating handle. It is not necessary to force or pull the material through the rollers and only a slight pressure should be required to keep the sheet drawing through the roller. *(See Figure 34.)*

49. HAND BRAKE.

Inasmuch as bent-up sheet stock is employed extensively in the repair of longerons, spars, frames, and ribs, the use of a hand brake will be found particularly desirable. By means of a hand brake, accurate bending of sheet stock may be accomplished readily. Most brakes accommodate wide sections, but the length of these sections is limited by the capacity of the hand brake. The hand brake consists essentially of three parts; the bed, the jaw, and the table. The bed is stationary and has a level drop surface. The jaw can be moved up and down by hand levers and can be adjusted in the horizontal plane by means of adjusting screws. The table is hinged to the bed in such a manner that it is possible to move the table through an arc of about 135 degrees. The horizontal adjustment of the jaw depends upon the thickness of the material being bent and the radius of the bend desired. It is important that this adjustment be made properly, as otherwise the material being bent may be seriously damaged. If it is desired to make a true radius inside the bend, the use of scrap material placed under the material between the clamping jaws will provide greater radius as required. Scratches should be avoided by protecting the sheets with wrapping paper. When utilizing this method of bending, adhere to the radii outlined in a preceding paragraph. *(See Figure 35.)*

Figure 35—Hand Brake

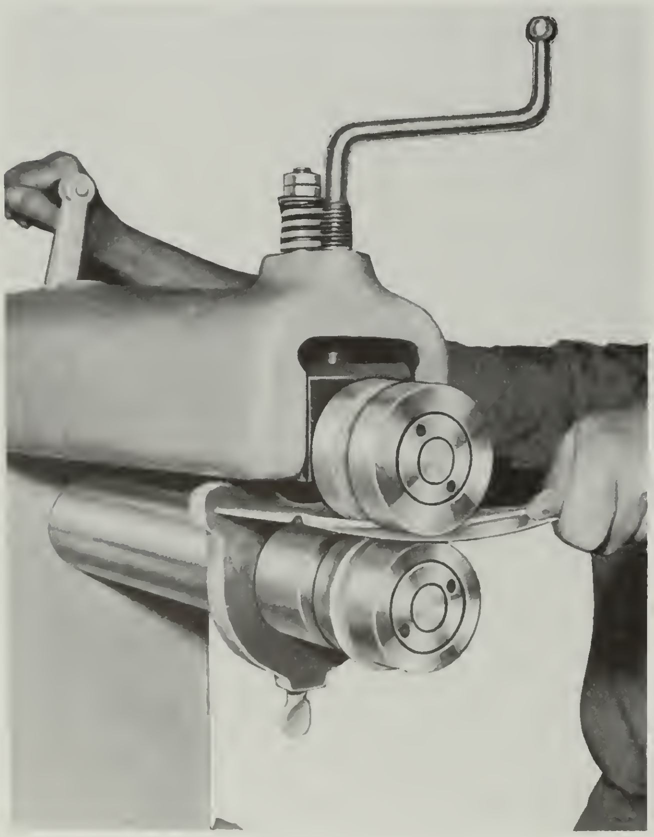


Figure 34—Hand Roller

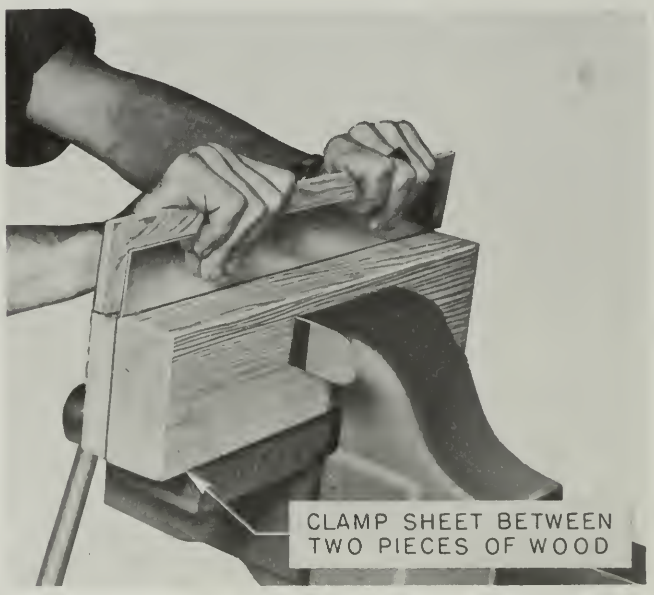


Figure 36—Bending Sheet Stock Locally

50. HAND METHODS OF BENDING.

Hand methods of bending are seldom as satisfactory as machine methods; however, if the work is done carefully, satisfactory results will be obtained. An improvised bending form may be made by clamping a steel bar or a hardwood block on each side of the material and forcing the extended flange down with a smooth board. *(See Figure 36.)* Where the material is too thick to bend in this manner, bend the material down with a steady pressure and, at the same time, strike the back of the board with a lead mallet. Narrow flanges are best bent by clamping the flange between the metal bars or wooden blocks and bending the main body of the material. If a sharper bend is desired, a lead mallet may be used to force the sheet against the block.

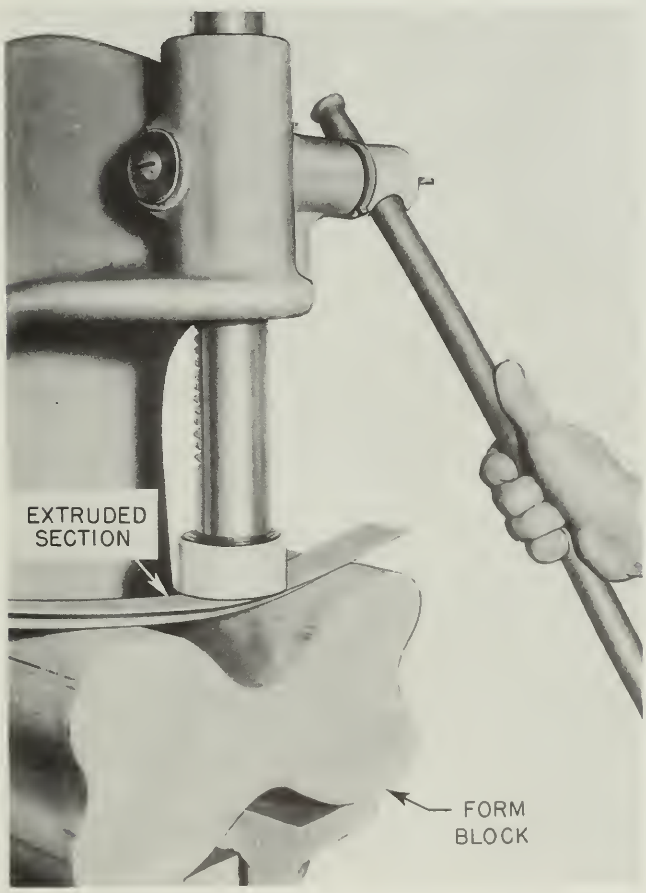


Figure 37—Arbor Press

51. ARBOR PRESS.

Where curved extruded sections are needed in repair, the use of an arbor press will greatly facilitate forming operations. Employing a C-type cast structure with a geared thrust arm, the arbor press, used in conjunction with various types of forming blocks, gives controlled thrust and a wide area of pressure. The arbor press has a variety of uses, but the bending of extruded sections is of primary concern. To set up and operate the machine properly, center the wood or lead forming block on the base, place the extruded section on the forming block, place a protective wood block over the extrusions, and apply a desired pressure by means of the hand lever. *(See Figure 37.)*

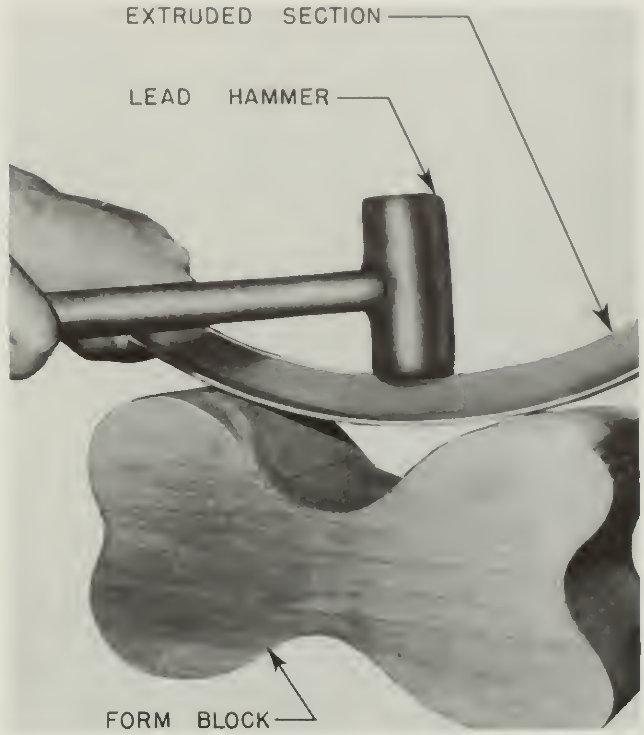


Figure 38—Forming Block

52. FORMING BLOCKS.

Forming blocks of desired size and shape may be fabricated from hardwood or lead to facilitate hand forming of extruded sections or bar stock. Used in conjunction with a lead mallet, such blocks assist in making various bends. Center the aluminum stock on forming block and strike with lead mallet in consecutive blows until correct curvature is obtained. Although this method is not as satisfactory as the machine methods, satisfactory bends may be made if care is taken. *(See Figure 38.)*

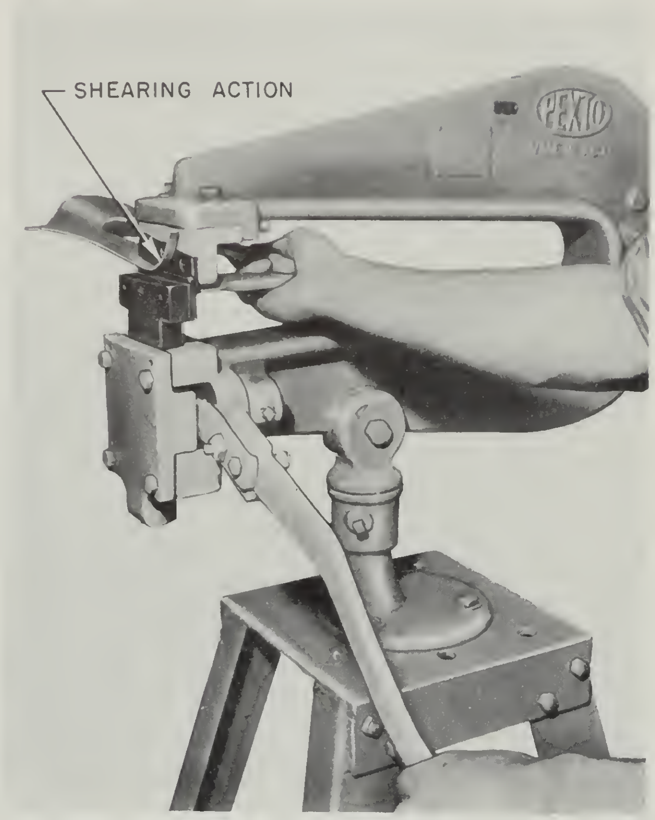


Figure 39—Trimming Machine

Figure 40—Use of Spiral Reamer

53. TRIMMING MACHINE.

Occasionally, straight shears will not suffice for special trimming or cutting operation. In such instances, the use of a trimming machine, made up of two cutting blades and a large C-shaped casting, is quite helpful. The shears may accommodate large pieces of sheet stock. Sheet material between the shear pieces is trimmed by actuating the lever handle. Successive cuts may be obtained by raising the lever handle, sliding material to new position, and again actuating lever handle. *(See Figure 39.)*

54. SPIRAL REAMER.

A spiral reamer used with a drill motor is quite satisfactory as a linear cutter for cleaning holes in alclad prior to patching. Mark the outline to be cut with a soft pencil and drill a 1/4-inch diameter hole in one corner. Insert the reamer into the hole, tilt it toward the operator, and cut on the line. Care should be taken to radius all corners. Sharp corners result in concentrations of stress and ultimate cracking of the material. *(See Figure 40.)*

55. STANDARD POWER AND HAND TOOLS.

The following list is a summation of standard power and hand tools that may be required in repair. The reference numbers refer to the tools illustrated on Figure 41.

REF. NO.	TOOL	USE
1	FILE HOLDER	USED TO HOLD FILES IN CURVED POSITION.
2	KEY-SEATING	USED TO KEY-SEAT DRILLED HOLES PRIOR TO INSTALLATION OF RIVNUTS.
3	SQUEEZER, RIVNUT	USED TO SQUEEZE RIVNUTS IN DRILLED AND KEY-SEATED HOLES FOR DE-ICER BOOT SCREWS.

REF. NO.	TOOL	USE
4	PIN PUSHER	USED TO PLACE DE-ICER PINS IN POSITION WHEN INSTALLING DE-ICER BOOT.
5	BURRING TOOL	USED TO SMOOTH TRIMMED EDGES OF METAL.
6	BURNISHING TOOL	USED TO SMOOTH MINOR SCRATCHES IN METAL.
7	PLIERS, CRIMPING	USED TO CRIMP TERMINAL LUGS ON ENDS OF ELECTRIC WIRES AND CABLES. TWO LARGER SIZES AVAILABLE.
8	IRON, SOLDERING	USED IN ELECTRICAL REPAIR WORK FOR SOLDERING TERMINALS AND CONNECTIONS, AND FOR GENERAL TINNING. VARIOUS SIZES, TIPS, AND WATTAGE AVAILABLE.
9	BRUSH, WIRE	USED TO CLEAN FILES, WELDS, AND RUSTY METAL AND FOR GENERAL ROUGHING.
10	CLAMP, "C"	USED FOR HOLDING TWO OR MORE PIECES OF METAL UNTIL DRILLING AND/OR RIVETING IS ACCOMPLISHED.
11	PLIER, CLAMPING	
12	SKIN FASTENER, CLECO	INSERTED IN DRILLED HOLES TO HOLD TWO OR MORE PIECES OF METAL UNTIL RIVETING IS ACCOMPLISHED.
13	GUN, CLECO SAFETY	USED TO INSERT SKIN FASTENERS.
14	WRENCH, TAP	USED IN CONJUNCTION WITH TAPS, EZY-OUTS, HAND AND TAPERED REAMERS.
15	PUNCH, DRIVE PIN "DRIFTS"	USED FOR DRIVING OUT PINS, RIVETS, AND BOLTS. USE NEXT SIZE SMALLER THAN HOLE.
16	CHISEL	USED FOR METAL CUTTING, AND REMOVAL OF RIVET HEADS WHERE DRILLING IS IMPRACTICAL.
17	PUNCHES, CENTER	USED FOR CENTERING HOLES AND AS A PUNCH BEFORE DRILLING TO ENSURE AN ACCURATE DRILL START. TYPES SHOWN ARE BLUNT AND SLIM TAPER.
18	FASTENERS, SKIN	USED TO SECURE METAL WHEN RIVETING, DRILLING, ETC.
19	CLAMPS, SKIN	
20	SAWS, HOLE	USED WITH PILOT IN HAND AND POWER DRILLS, THESE TOOLS ARE EXCELLENT FOR CUTTING CIRCULAR HOLES IN DAMAGED SHEET METAL PRIOR TO APPLICATION OF STANDARD BUTTON PATCH. SIZES RANGE FROM 5/8 TO 2 INCHES.
21	PILOT, HOLE SAW	USED TO PILOT HOLE SAWS.
22	WHEELS, EMERY	SMOOTH AND ENLARGE INSIDE EDGES OF LARGE HOLES. SIZES SHOWN 1-1/2 AND 2 INCHES.

REF. NO.	TOOL	USE
23	DRILL, TWIST	THIS TYPE DRILL USED WITH HAND OR POWER DRILLS IS STANDARD. SIZES RANGE FROM NO. 80 TO 2 INCHES.
24	REAMER, HAND, SQUARE SHANK, SPIRAL FLUTED	USED FOR ENLARGING DRILLED HOLES TO DESIRED SIZE. VARIOUS SIZES AVAILABLE.
25	REAMER, SQUARE SHANK, TAPER PIN, SPIRAL FLUTED	USED FOR ENLARGING DRILLED HOLES IN SHEET STOCK AND FOR REAMING SHAFT HOLES FOR TAPER PINS. IN ELECTRIC AND AIR MOTORS, REAMERS ARE EXCELLENT FOR MAKING CUTOUTS IN ALUMINUM MATERIAL.
26	EXTRACTOR, EZY-OUT	USED FOR REMOVING THE REMAINING PORTIONS OF BROKEN SCREWS, BOLTS, AND PINS. DRILL HOLE IN OBJECT APPROXIMATELY, ONE-HALF THE DIAMETER , INSERT EZY-OUT, AND WITHDRAW OBJECT BY COUNTER-CLOCKWISE ROTATION.
27	TAP	USED FOR THREADING VARIOUS SIZES OF HOLES FOR ALL STANDARD THREADS. AVAILABLE IN DIFFERENT DIAMETERS, THREADS PER INCH, AND PITCH.
28	COUNTERSINK	USED FOR PREPARING HOLES IN HEAVY STOCK FOR USE OF COUNTERSUNK RIVETS AND SCREWS. AVAILABLE FROM 1/8-TO 2 INCHES. DEGREE OF TAPER FOR RIVETS AND SCREWS, 78°, 82°, AND 100°.
29	COUNTERSINK, BACK	USED FOR COUNTERSINKING IN LOCATIONS WHERE MOTIVE POWER CANNOT BE APPLIED ON SIDE DESIRED.
30	SPOT-FACER, BACK	USED WITH VARIOUS SIZED PILOTS WHERE ACCURACY OF HOLE LOCATION IS DESIRED. ALSO USED FOR FLUSHING BOLT HEADS IN CASTINGS. TYPE SHOWN IS USED IN LOCATIONS WHERE MOTIVE POWER CANNOT BE APPLIED ON SIDE DESIRED, BUT STRAIGHT SHANK TYPE ALSO IS AVAILABLE.
31	FILES, HAND	GENERAL: USED FOR FINISHING WORK ON ALL TYPES OF METAL, PLASTIC, AND FIBER, AND ENLARGING HOLES, BURRING EDGES, SMOOTHING CUTS, REMOVING METAL, TOOL SHARPENING, ETC. MANY DIFFERENT LENGTHS, CUTS, AND SECTIONS ARE OBTAINABLE. VIXEN: A VERY COARSE FILE USED ON ROUGH WORK REQUIRING REMOVAL OF LARGE QUANTITIES OF MATERIAL. BASTARD: A COARSE FILE AVAILABLE IN VARIOUS LENGTHS AND SECTIONS. MILL: A FLAT FINE-CUT FILE FOR FINAL FINISHING AND TOOL SHARPENING. SMOOTH: SIMILAR TO MILL, BUT OBTAINABLE IN VARIOUS SECTIONS. SWISS NEEDLE: A FINE, SLIM, TAPERED FILE USED ON DELICATE WORK AND IN RESTRICTED AREAS.

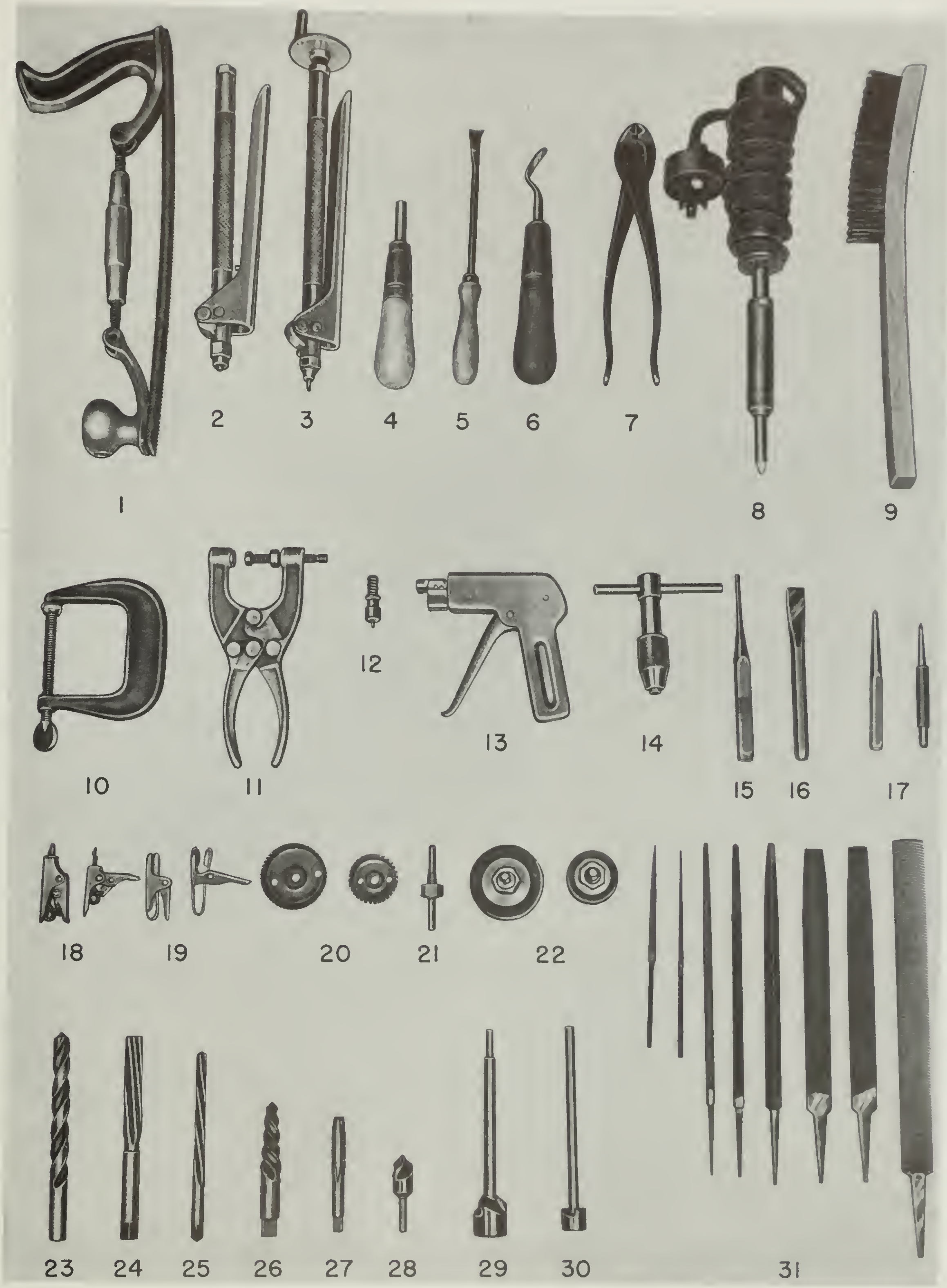


Figure 41—Standard Power and Hand Tools

56. STANDARD METAL-WORKING HAND TOOLS.

The following list is a brief summation of hand tools that may be required in repair. The reference numbers refer to the tools illustrated on Figure 42.

REF. NO.	TOOL	USE
1	SNIPS, TINNERS	USED FOR STRAIGHT LINE CUTTING OF SHEET METAL UP TO .064" THICK.
2	SNIPS, COMBINATION CIRCLE "DUCK BILL"	USED FOR STRAIGHT LINE AND CURVED CUTTING.
3	SNIPS, LEFT-HAND, DOUBLE ACTION "DUTCHMANS"	USED FOR CUTTING TO LEFT AND/OR CUTTING ENDS OF TUBING.
4	SNIPS, RIGHT-HAND, DOUBLE ACTION "DUTCHMANS"	USED FOR CUTTING TO RIGHT AND/OR CUTTING ENDS OF TUBING.
5	HACK SAW, LARGE, ADJUSTABLE	USED FOR METAL, FIBER, AND PLASTIC CUTTING. BLADES VARY IN LENGTH AND IN NUMBER OF TEETH PER INCH.
6	HACK SAW, SMALL	USED FOR LIGHT CUTTING OF METAL, FIBER, OR PLASTIC.
7	HACK SAW, KEYHOLE	USED FOR CUTTING HOLES IN RESTRICTED PLACES.
8	VISE, DRILL PRESS, SPEED	USED TO HOLD OBJECTS WHILE DRILLING.
9	DRILL, HAND	USED FOR RESTRICTED OR SLOW DRILLING, AND ALSO IN THE PRESENCE OF GAS FUMES, IF AN AIR MOTOR IS NOT AVAILABLE.
10	DIVIDERS	USED FOR SHEET METAL LAYOUT.
11	SCRIBE	
12	MALLET, LEAD	USED TO FORM AND BEND METAL.
13	MALLET, RUBBER	
14	MALLET, RAWHIDE	
15	MALLET, PARALYN	
16	MALLET, PARALYN	
17	HAMMER, FINISHING	USED TO FORM AND FINISH METAL.
18	RULE, SQUARE, CENTER AND PROTRACTOR COMBINATION	USED FOR SHEET METAL LAYOUT.
19	RULE, 6" RIGID	
20	PROTRACTOR, FLAT	
21	CALIPERS, SLIDE	ACCURATE INSIDE OR OUTSIDE DIAMETER MEASUREMENT IS OBTAINED WITH THIS TOOL.
22	RULE, 6' FLEXIBLE	USED FOR SHEET METAL LAYOUT.

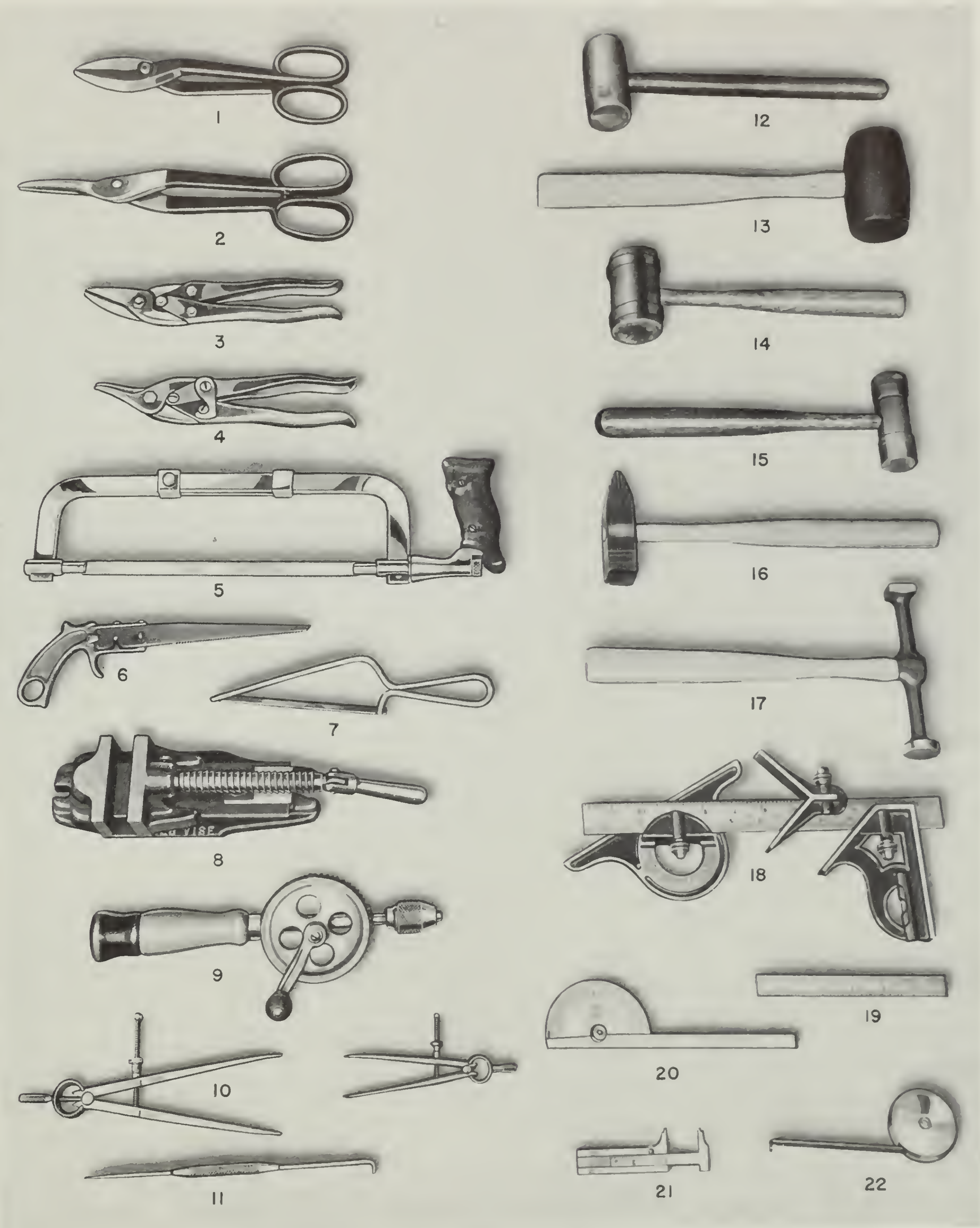


Figure 42—Standard Metal-working Hand Tools

57. ALUMINUM ALLOY MATERIAL SPECIFICATION EQUIVALENTS.

MATERIAL		U.S.A. SPEC.	BRITISH SPEC.	BRITISH REPLACEMENT	U.S.A. TENSILE
2S	SHEET	QQ-A-561	L17, L16, L14		
2S	TUBE	WW-T-783	T9		7-10 TONS
2S	ROD	QQ-A-411	L34	L44	
3S	SHEET	QQ-A-359			6.5 TONS
14ST	FORG.	QQ-A-367			32.5 TONS
A17S	RIVET	25526	DTD 327		
17ST	BAR	QQ-A-351	L1		24.5 TONS
17ST	FORG.	QQ-A-367	L1		27.5 TONS
17ST	TUBE	WW-T-786	T4		24.5 TONS
24SO	SHEET	QQ-A-355	DTD 270	DTD 351	15.5 TONS
24SO	ALCLAD	QQ-A-362	DTD 275	DTD 390	14 TONS
24ST	SHEET	QQ-A-355	DTD 270	DTD 351	28 TONS
24ST	ALCLAD	QQ-A-362	DTD 275	DTD 390	25 TONS
52S	SHEET	QQ-A-318	DTD 266	L44 OR L38	17.5 TONS
52S	TUBE	57-187-3		DTD 310A	17.5 TONS
195-T6	CAST.	AN-QQ-A-390	DTD 298		16 TONS
356-T6	CAST.	11308		L33	15 TONS
DIE CAST.		AN-QQ-A-366		DTD 245	16 TONS
SAND CAST. NO. 43		11311		DTD 133B	8.5 TONS

58. STEEL ALLOY MATERIAL SPECIFICATION EQUIVALENTS.

MATERIAL	U.S.A. SPEC.	BRITISH SPEC.	BRITISH REPLACEMENT	U.S.A. TENSILE OR REMARKS
1020 SHEET	AN-QQ-S-651	S84	S3	
1020 TUBE	AN-QQ-T-846	T26		
1020 ROD	AN-QQ-S-646	S21, S14		
1025 SHEET	AN-QQ-S-651	S3		
1025 TUBE			T1	
1025 ROD	AN-WW-T-846		S1	
1095 SHEET	AN-QQ-S-666			
2315 ROD		S15		
2330 ROD	AN-QQ-S-689		S69, S2	70 TONS
2335 ROD		S69		
X4130 SHEET	AN-QQ-S-685		DTD 124	70 TONS
X4130 TUBE	57-180-2	T45, DTD 347	195, 178	T35 SMALL
X4130 ROD	AN-QQ-S-684		S2	70 TONS
4130 ROD	AN-QQ-S-684		S2	70 TONS
4140 BAR	10083			90 TONS
AISI 302 SHEET	AN-QQ-S-772		DTD 171	40 TONS
AISI 302 SHEET	AN-QQ-S-772		DTD 166	55 TONS
AISI 302 SHEET	AN-QQ-S-772			67 TONS
AISI 302 SHEET	AN-QQ-S-772			78 TONS
AISI 302 SHEET	AN-QQ-S-772			83 TONS
AISI 303 BAR			DTD 171	35 TONS
AISI 347 SHEET	AN-QQ-S-757		DTD 166	45 TONS
BARS			M.S.	1010 NON-STRUCTURAL
SS TUBES			203A, 211	

59. COPPER ALLOY MATERIAL SPECIFICATION EQUIVALENTS.

MATERIAL	U.S.A. SPEC.	BRITISH SPEC.	BRITISH REPLACEMENT	U.S.A. TENSILE OR REMARKS
COPPER SHEET	QQ-C-501		B.S. 444, 518	B.S. 176 BONDING OR 125 BUS BAR
COPPER TUBE	WW-T-799		T7, T51	
BRASS SHEET	QQ-B-636		B.S. 265, 266	
BRASS TUBE	WW-T-791		T8	
BRASS BAR	QQ-B-636		B.S. 251	
BRONZE SHEET	11065		B.S. 407	
BRONZE EVERDUR			DTD 312	(DTD 265A)
BRONZE SAE 62			B.S. 383	
BRONZE SAE 64			B8	
BRONZE SAE 43	QQ-B-726		B.S. 208	
MN BRONZE CAST.	QQ-B-726		B.S. 208	
MN BRONZE BAR	QQ-B-721			25-50 TONS
SILICON BRASS	QQ-C-591		DTD 263, 267	
BRASS FORG. A	QQ-B-611		B.S. 218	60/40 FORG.
BRASS BAR B	QQ-B-611		B.S. 249	60/40 MACH.
BRASS BAR C	QQ-B-611		B.S. 265, 266	65/35 BAR
BRASS BAR D	QQ-B-611			64/36 MACH.
BRASS SHEET E	QQ-B-611		B.S. 267	70/30 SHEET
BRASS CAST.	QQ-B-621		B.S. 383	
PHOSPHOR BRONZE	QQ-B-746			
BRONZE CAST.	QQ-B-691		B.S. 383	10 COMP. CU-SN-ZN
BRONZE CORED	11306		B8	4 COMP. CU-SN-ZN
AL BRONZE CAST.	QQ-B-671		DTD 174	
MN BRONZE CAST.	QQ-B-721		B.S. 250	

60. WIRE AND CABLE MATERIAL SPECIFICATION EQUIVALENTS.

MATERIAL	U.S.A. SPEC.	BRITISH SPEC.	BRITISH REPLACEMENT	U.S.A. TENSILE OR REMARKS
7 X 7 FLEX.	48-22	5W2		TINNED CABLE
7 X 7 EX. FLEX.	48-36	DTD 181		STAINLESS STEEL
7 X 19 EX. FLEX.	48-36	DTD 181		STAINLESS STEEL
SPRING WIRE	48-26	DTD 239		STEEL 100-150 TONS
SPRING WIRE	48-7		DTD 239	STEEL
1085 GRADE	48-7		DTD 239	85-125 TONS
1095 GRADE	48-7		DTD 239	NO TENSILE
1350 GRADE	48-7		DTD 239	70-115 TONS
6150 GRADE	48-7		DTD 4	NO TENSILE
STEEL WIRE	48-7		M.S.	
S.S. WIRE	48-37			18/8 100-135 TONS
STEEL WIRE	48-24		DTD 215	95-137 TONS
SOFT CU WIRE	QQ-W-341			16-18 TONS

SECTION 2
FUSELAGE

2

1. FUSELAGE GENERAL CONSTRUCTION.

The complete fuselage is comprised of three separate detachable sections consisting of the engine mount, the front fuselage section, and the rear fuselage section. The entire fuselage has a maximum cross-sectional height of 76-1/2 inches and a maximum width of 46-13/16 inches. The detachable engine mount and the front fuselage structure from the firewall to a position aft of the rear cockpit seat are of chrome molybdenum steel construction with welded steel joints and fittings. On the earlier AT-6C Series Airplanes and previous North American Aviation, Inc., trainer aircraft, the fuselage section from the rear cockpit to the tail is of riveted 24ST alclad aluminum semimonocoque construction. On the later AT-6C Series Airplanes, the rear fuselage section is constructed of wood and the entire structure is assembled with casein glue. The removable 24ST alclad aluminum alloy side panels of the front fuselage section serve only as fairings and do not carry loads. The term "fuselage station" used in this Section and on NAA production drawings specifies the location of tubing cluster joints and formers measured in inches aft of the firewall *(see Figure 1)*. Also, on production drawings, tubing cluster joints of the front fuselage are numbered from the firewall consecutively in reference with a U or L to designate an upper or a lower tubing cluster joint *(see Figure 1)*. The complete fuselage is attached to the upper surface of the centersection by bolts passing through the welded longeron fittings.

2. CONSTRUCTION OF ENGINE MOUNT AND FRONT FUSELAGE TRUSS.

The engine mount and the front fuselage truss extending from the firewall to a point just aft of the rear cockpit are constructed of various sizes of chrome molybdenum steel tubing conforming to SAE Spec. X4130 *(see Figure 2)*. The steel tube truss-type engine mount is

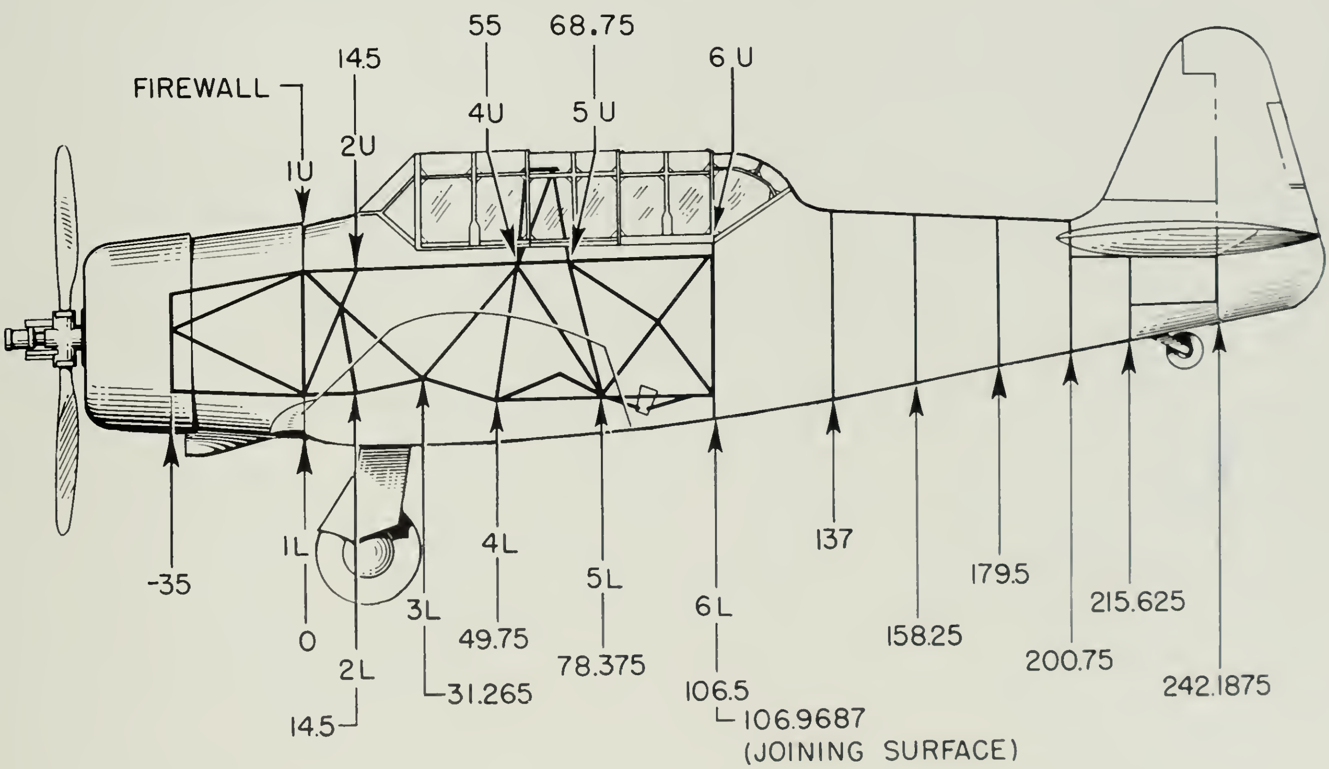


Figure 1—Fuselage Stations

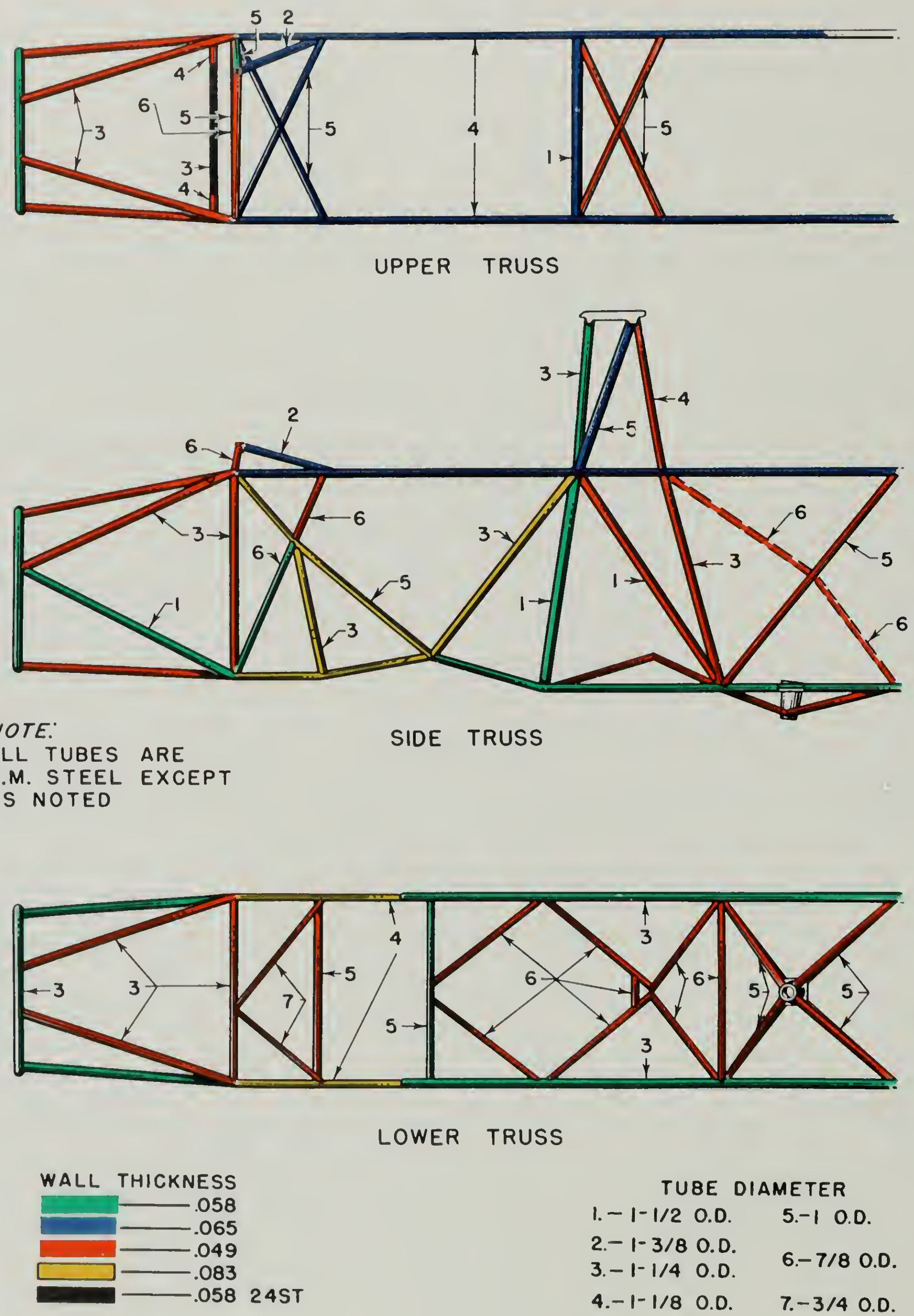


Figure 2—Engine Mount and Front Fuselage Tube Sizes

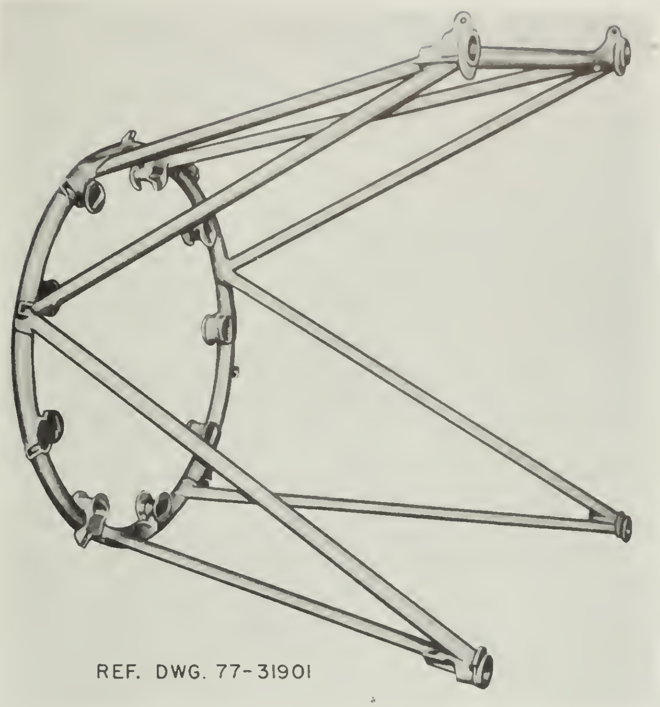


Figure 3—Engine Mount Truss Assembly

bolted to the two upper and the two lower longerons of the front fuselage tubing truss by single-bolt fittings. The engine is attached to the engine mounting ring in nine places by rubber-cushioned, dynamic mount fittings *(see Figure 3)*. Inasmuch as the cross bracing in the lower fuselage truss is discontinuous between the centersection attachment points *(see Figure 4)*, this fuselage truss structure is rigid only when attached to the centersection, and care should be taken to support this tubular framework properly when it is removed from the centersection. This type of truss is in accordance with conventional practice. The firewall is not considered a structural member although it actually gives some lateral stability to the front diagonal tube. It is to be noted that no welding is done on any tubes between their extremities. This provides ample space on the tubes for clamping or otherwise attaching the equipment accessories and other components. Reinforcing plates are welded on some joint clusters *(see Figure 5)*. Electric arc welding is employed throughout the entire structure in the original construction, but oxyacetylene welding may be substituted in making repairs if arc welding facilities are not available. All tube sizes and wall thicknesses are noted on the truss assembly diagram *(see Figure 2)*.

3. WELDING STEEL PARTS - GENERAL.

The process of joining steel parts by welding consists in fusing the metal of the welding wire or rod with the metal of the joint ends until the joint is built up with new metal. This process can be accomplished by either electric arc welding or by oxyacetylene welding. The fused metal of the joint is of a cast structure and does not have the physical properties or strength of the metal parts or the welding wire before being fused. The sections of the tube adjoining a gas weld will be annealed by the welding heat for a distance of from 1/4-to 3/4-inch on each side of the weld. In cases of electric welding, this distance will be greatly reduced,as the weld proceeds much faster than in the case of gas welding, and no preheating of the material is necessary. The melting point of steel is approximately 1371°C (2500°F) and the welder's torch flame is near this point at all times. A combination eye and face shield and a leather apron should be worn while welding, and proper ventilation should be provided. Inasmuch as the presence of gas vapors will be a fire hazard all necessary fire precautions must be taken before any welding is attempted on all damaged ships that have been in service. Sandpaper or use a wire brush and clean all affected areas prior to welding if a completely new weld is to be made. If a weld is to be made over a failure in an electric weld bead, chip and file off all of the existing bead before applying a new weld to the area. After a weld is made, do not file or smooth the weld or apply any solder or other filler to improve the appearance of the weld.

4. ELECTRIC ARC WELDING OF STEEL.

Before starting to weld, make certain that the surface of the parts to be welded is free of loose scales, oxides, oil, and foreign matter. Jigs, clamping devices, and tack welding shall be used wherever required to control warping and ensure proper alignment. Preheating, scarfing, and chamfering are not necessary on any tubing of the front fuselage truss, as none of the tubes are over .083 inch thick. Preheating is required only on heavy fittings and forgings, and chamfering is required only on material of .140-inch thickness and greater. The proper electric arc-welding rods to be used with chrome molybdenum steel shall be in accordance with the following table *(see Figure 7)*. In general, a heavily coated, mild steel welding rod should be used, and the diameter of

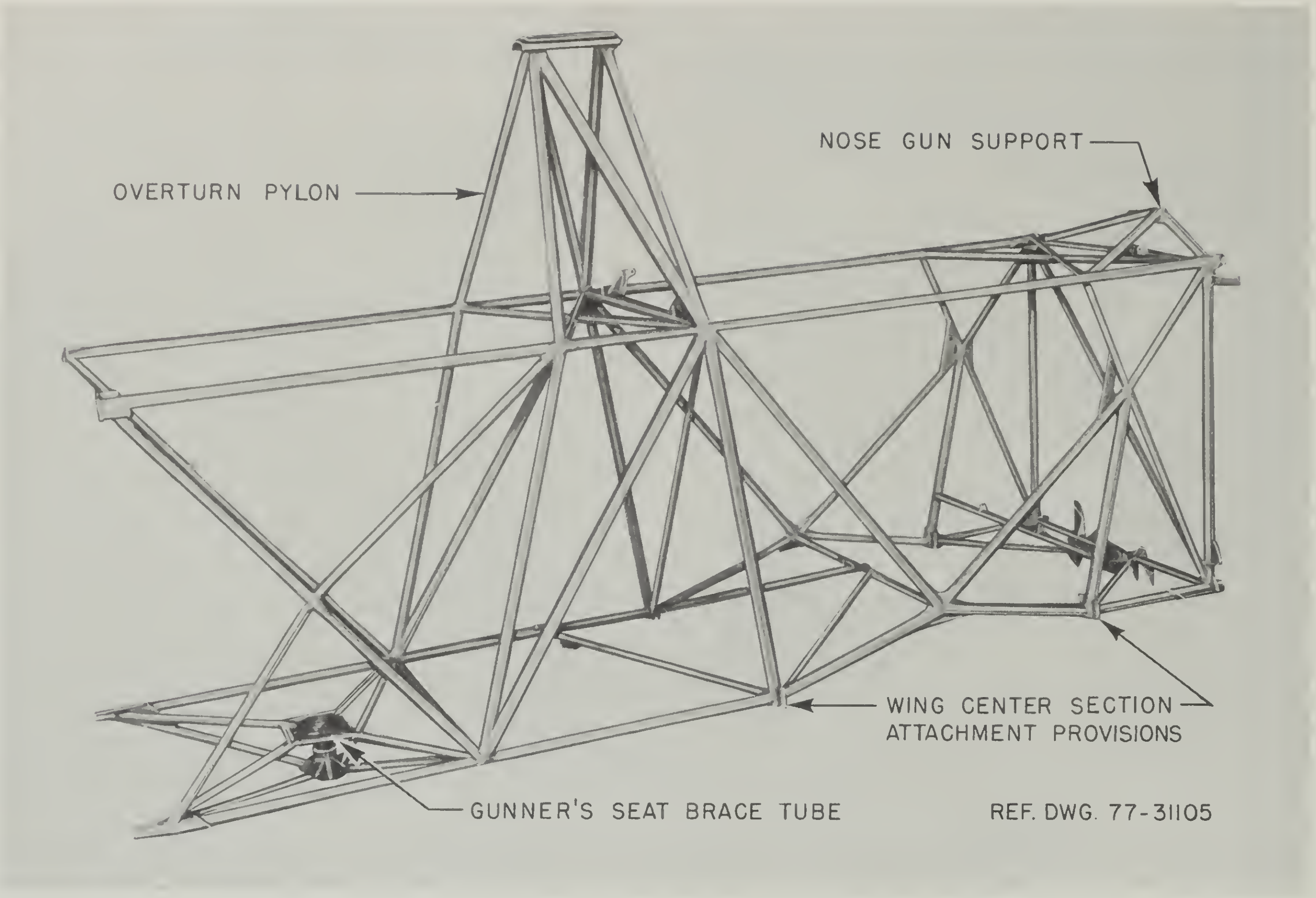


Figure 4—Front Fuselage Truss Assembly

the electrode should not be greater than the tube wall thickness unless the operator increases the travel speed sufficiently to prevent overheating, undercutting, and burning through the metal. The length of the arc should be held to approximately 1/8-inch. Polarity shall be as recommended by the electrode manufacturer or as found suitable for the specific work being accomplished. All tack welding and all weld endings shall be accomplished with the use of a crater eliminator if one is available *(see Figure 6)*. The small hole or crater at the weld end creates a stress concentration and a fatigue point, and should therefore be eliminated if possible. The use of a crater eliminator will alleviate this condition and a smooth weld end will result *(see Figure 8)*. Avoid rewelding as porosity in the weld may result. When a weld is built up by two or more beads or passes, the preceding weld must be cleaned free of scale or flux by chipping or scraping and followed by brushing with a wire brush. Welding shall not be dressed by removing metal from the joint unless further welding is to be done on the dressed region. Unless otherwise specified, the maximum width of welds for material thickness of not over .040 inch shall be 1/4-inch *(see Figure 9)*. The depth of penetration shall be between 25 and 40 percent of the thickness of the base metal.

5. OXYACETYLENE WELDING OF STEEL.

The oxyacetylene process is still considered the most flexible type of welding and generally best suited for repair work on aircraft. However, both electric arc welding and oxyacetylene welding are acceptable, and repairs outlined will be applicable for either type of welding. Welding wire used for oxyacetylene welding of chrome molybdenum tubing must conform to Spec. QQ-W-351, Grade E. The torch tips should be of proper size for the

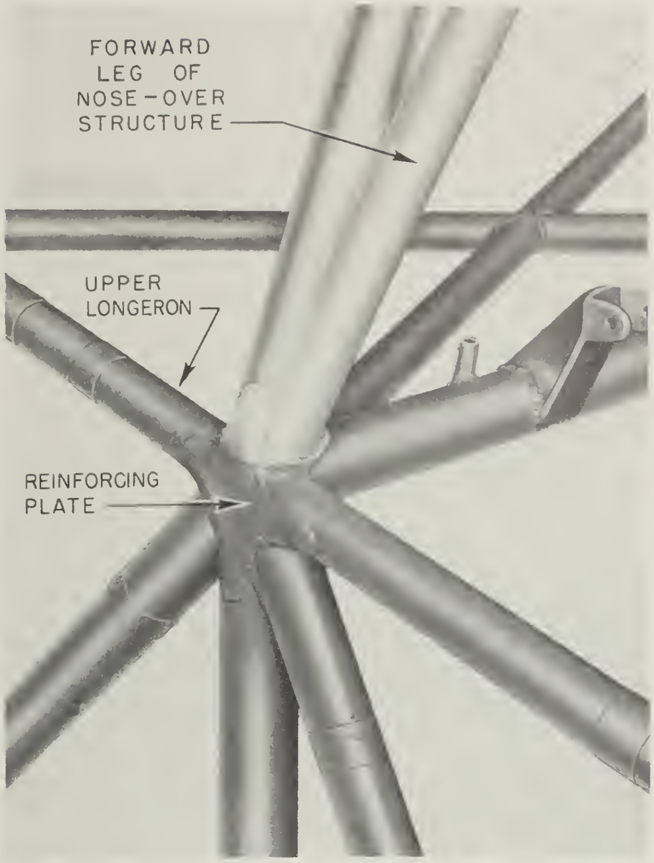


Figure 5—Upper Tube Cluster Joint at Station 55

thickness of the material to be welded. The following commonly used torch tip sizes are satisfactory:

THICKNESS OF STEEL	DIAMETER OF HOLE IN TIP	DRILL SIZE
.015 TO .031	.026	NO. 71
.031 TO .065	.031	NO. 68
.065 TO .125	.037	NO. 63
.125 TO .188	.042	NO. 58
.188 TO .250	.055	NO. 54
.250 TO .375	.067	NO. 51

For chrome molybdenum tubing of under .115-inch thickness, the welding rod should be a plain copper-coated carbon steel, 1/16- or 3/32-inch diameter rod; no flux is necessary. Preferably keep the flame pointed in the direction of welding in order to preheat the material. Maintain as neutral a flame as possible, for an excess of acetylene in the flame will carbonize the weld and an excess of oxygen will oxidize the weld. The feather part of the flame should not be more than 1-1/2 times the length of the flame cone and not more than 1/8-inch long. Avoid rewelding, as overheating and porosity may result in the weld. At the end of the weld do not raise the torch suddenly from the weld, as this action may cause a pinhole in the weld. Welds shall not be dressed by removing metal from the joint unless further welding is to be done on the dressed region. If the thinnest of the tubes to be welded is less than .040 inch, the maximum width of the weld should not exceed 1/4-inch. Standard oxyacetylene welding equipment should be used *(see Figure 10)*.

6. CONDITION OF COMPLETED WELDS.

The finished weld should incorporate the following characteristics:

a. The seam should be smooth and of uniform thickness.

b. The weld should be built up to provide extra thickness at the seam.

c. The weld metal should taper off smoothly into the base metal.

d. No oxide should be formed on the base metal at a distance of more than 1/2-inch from the weld.

e. The weld should show no signs of blow holes, porosity, or projecting globules.

f. The base metal should show no signs of pitting, burning, cracking, or distortion.

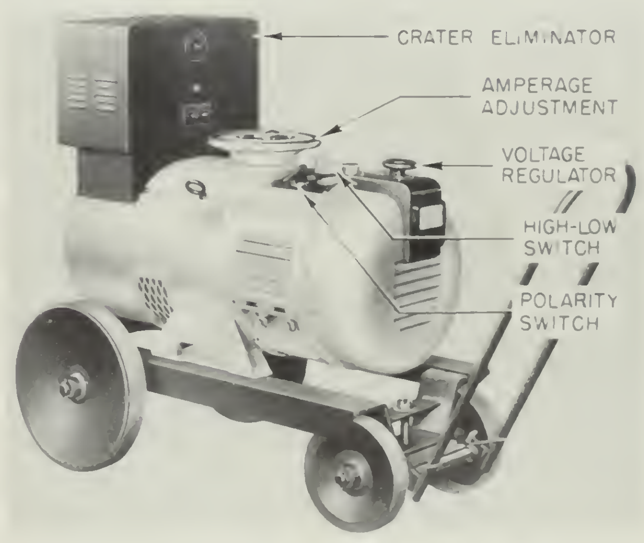


Figure 6--Arc Welding Generator With Crater Eliminator

BASE MATERIAL	THICKNESS IN INCHES	WELDING ROD MATERIAL	DIAMETER INCHES	SPECIFICATION	ARC VOLTAGE	AMPERAGE
CM STEEL (4130 AND X4130)	UP TO .064 .064 TO .078 .078 TO .093	PLAIN CARBON STEEL, HEAVILY COATED "WILSON 520", "G.E. W25", OR "AIRCO 90"	1/16 5/64 3/32	10286-A GR. 5-E	17-20 17-21 17-21	20-40 25-60 30-80

Figure 7—Arc Welding Electrode Requirements

g. The beads shall be of correct size and number.

h. The depth of penetration shall be sufficient to ensure fusion of base metal and filler rod.

i. Objectionable welding scale shall be removed by wire brushing or sandblasting.

7. GENERAL REPAIR OF STEEL TUBES.

Repairs to the forward fuselage truss and engine mount consist of smoothing small nicks, scratches, and dents; reinforcing cracked members; reinforcing dented areas; splicing damaged members; replacing damaged members when splicing is impractical; and correcting minor distortion. With the exception of correcting minor distortion, all repairs to the forward fuselage truss are accomplished by welding. No bolted-type splices nor reinforcements are allowable. Electric arc welding and oxyacetylene welding are acceptable, and the welding method used will be determined by the location of the damage and the available equipment. The nature of the damage will determine whether or not the engine or rear fuselage section must be removed from the fuselage truss. In general, smoothing small dents and correcting minor distortion can usually be accomplished without disassembly of the major components. The intense heat caused by all types of welding necessitates isolating the area to be welded from all parts or members which might be

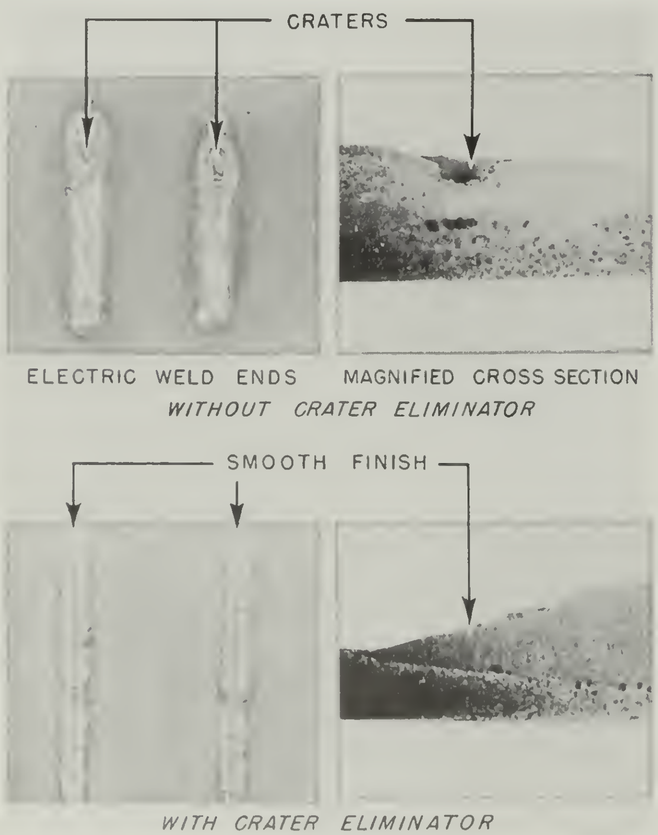


Figure 8—Typical Weld Ends

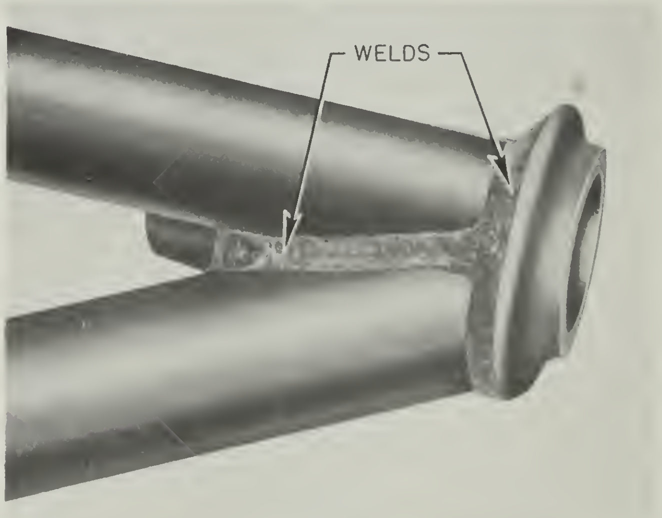


Figure 9—Typical Arc-Welded Engine Mount Joint

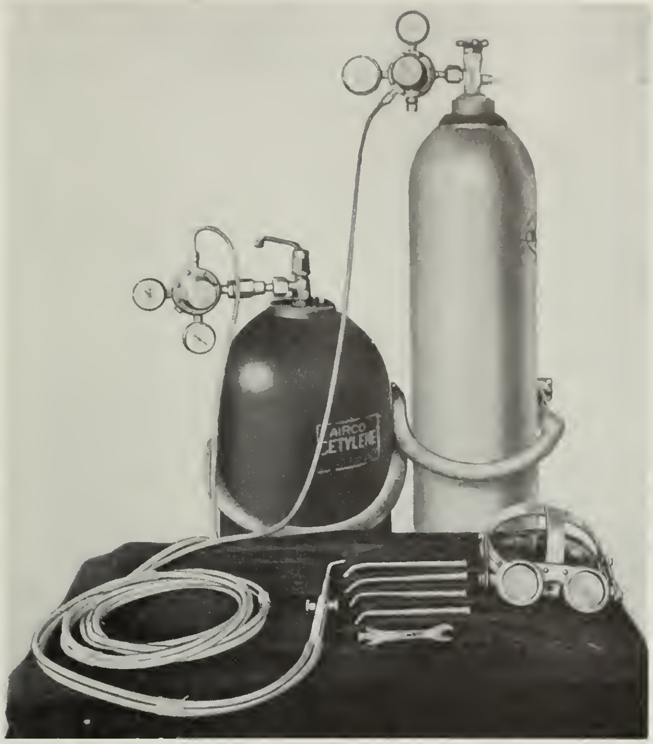


Figure 10—Oxyacetylene Welding Equipment

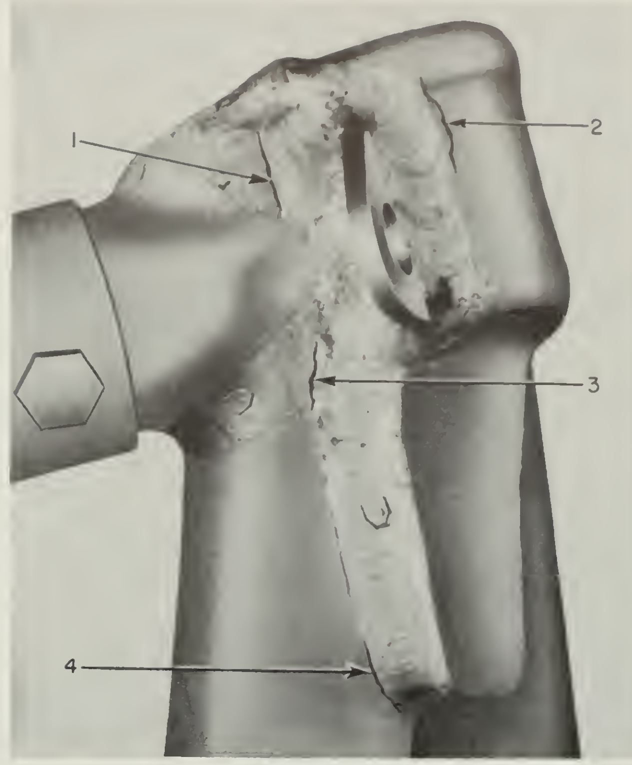


Figure 11—Typical Welding Failures

injured by contact with heat. Sheet asbestos or heavy wet rags will furnish sufficient insulation for adjacent furnishings. All fire precautions should be observed before any welding is started. All tubing or sheet stock used for repairs must be X4130 chrome molybdenum steel. Tubing used for telescope reinforcements or for splicing must be of the same wall thickness as or greater than that of the original member *(see Figure 2)*.

8. NEGLIGIBLE DAMAGE TO STEEL TUBES.

Some forms of damage to tubular structures may be considered negligible. Such damage may take the form of slight indentations, scratches, or minor bowing. Smooth dents not exceeding 1/20 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 scrutinized for the presence of sharp nicks and deep scratches which may have resulted from service. These nicks and scratches 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, fine emery paper, or steel wool. When this is accomplished, high concentrations of stress disappear.

9. ESTIMATING EXTENT OF DAMAGE TO STEEL TUBES.

When inspecting the forward fuselage truss and engine mount for possible damage, the structure surrounding any visual damage must be carefully scrutinized to ensure that no secondary damage remains undetected. Secondary damage may be produced in some structure remote from the location of the primary damage by the transmission of the damaging concentrated load over the normal course toward the basic reacting structure. Damage of this nature usually occurs where the most abrupt change in load travel is experienced. If this damage remains undetected, loads applied in the normal course of operation may cause failure of the part. Visually examine all joints for cracks, welding flaws, or failures *(see Figure 11)*. If any doubt exists as to the presence of cracks or other flaws, test the structure as outlined in an applicable subsequent paragraph.

10. SMOOTH DENTS IN STEEL TUBES.

A minor smooth dent in steel tubing may often be removed by the following procedure: Remove one of the self-tapping screws provided at the extremities of the main steel tubes and apply

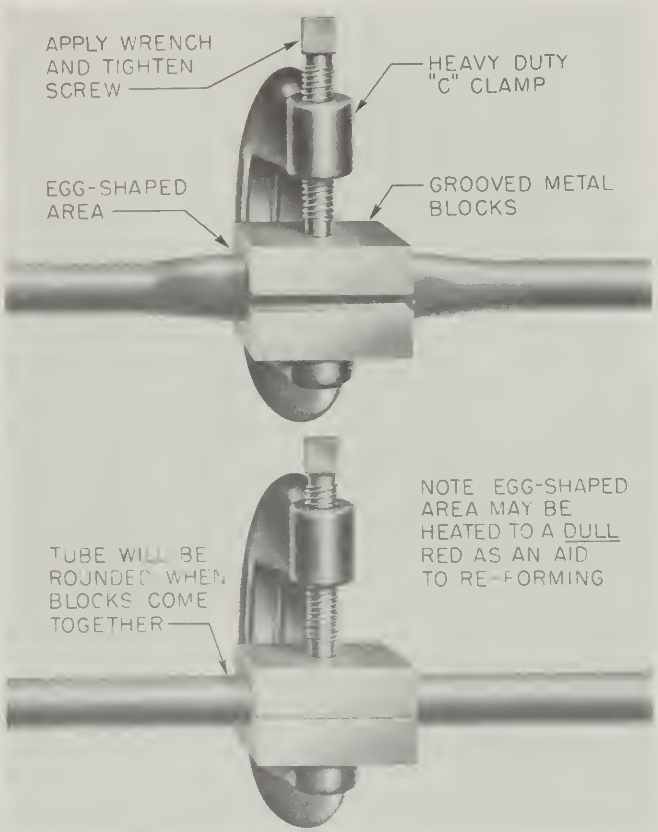


Figure 12—Correcting Oval-shaped Steel Tube Distortion

an air pressure of upwards of 75 lbs./sq.in. to the inside of the steel truss. Heat the dented area evenly to a dull red with an acetylene torch until the internal air pressure forces out the dent and restores the original tube contour. If internal air pressure and heat are not sufficient to remove the dent, tack weld a welding rod to the center of the dent and pull on the rod while heating the area. After the dent is removed, allow the area to air-cool and then release the internal air pressure. Replace the previously removed self-tapping screw.

Important: Do not apply heat over a dull red to the middle third of the length of any tube.

11. STEEL TUBE CIRCUMFERENCE BENT TO AN OVAL SHAPE.

Where the circumference of a steel tube is bent to an oval shape, the area may be restored to normal in the cold condition by pressure exerted on the area through grooved steel form blocks *(see Figure 12)*. Drill a steel block to the diameter of the damaged tube; then saw the block along the axis of the hole and separate the two sections of the block. Apply the two form block sections to the oval-shaped area on the affected tube. Slip a heavy clamp over the blocks, tighten the clamp, and exert pressure on the area until the oval-shaped tube area is restored to the normal circular shape *(see Figure 12)*. If difficulty is encountered in shaping the tube in the cold condition, heat the area to a dull red; then apply the steel blocks and clamp. Remove the clamp and the blocks. If the oval-shaped area is longer than the length of the steel form blocks, reapply the form blocks and the clamp to successive affected areas until the entire length of the oval-shaped area is restored to the normal circular shape.

12. BOWED STEEL TUBES.

Steel tubes which have been bowed without evidence of cracking may be straightened in the cold condition as shown *(see Figure 13)*. Cut three hardwood blocks grooved to fit the contour of the tube, and line the grooves with leather or canvas. Obtain a length of channel iron equal to the length of the bow in the tube. Locate one of the grooved wood blocks at either extremity of the bow and apply the channel iron beam so that the beam spans the bowed area and backs up the two blocks. Apply the third block on the opposite side of the tube at the point of the maximum bend near the center of the bow. Slip one end of a heavy "C" clamp over the channel iron beam and tighten the clamp down on the block at the center of the bend. In order to allow for springback of the tube, continue tightening the clamp until the tube is bent slightly in the opposite direction *(see Figure 13)*. Remove the clamp and the blocks. Check the alignment of the tube by placing an accurate straight edge on both the side and the top of the tube. If the straight edge check reveals a slight bow in the tube, reapply the blocks and the clamp and check until the tube lines up with a straight edge in both reference planes. The maximum allowable tube bow is 1/16-inch measured from the straight edge to the outer circumference of the tube. If the tube is free from cracks and is within the allowable maximum bow of 1/16-inch after straightening, the tube is perfectly acceptable without additional reinforcement. However, if cracks appear at the point where the maximum bow was corrected, drill a No. 40 hole at the ends of the crack and weld a split steel sleeve over the area as outlined in the

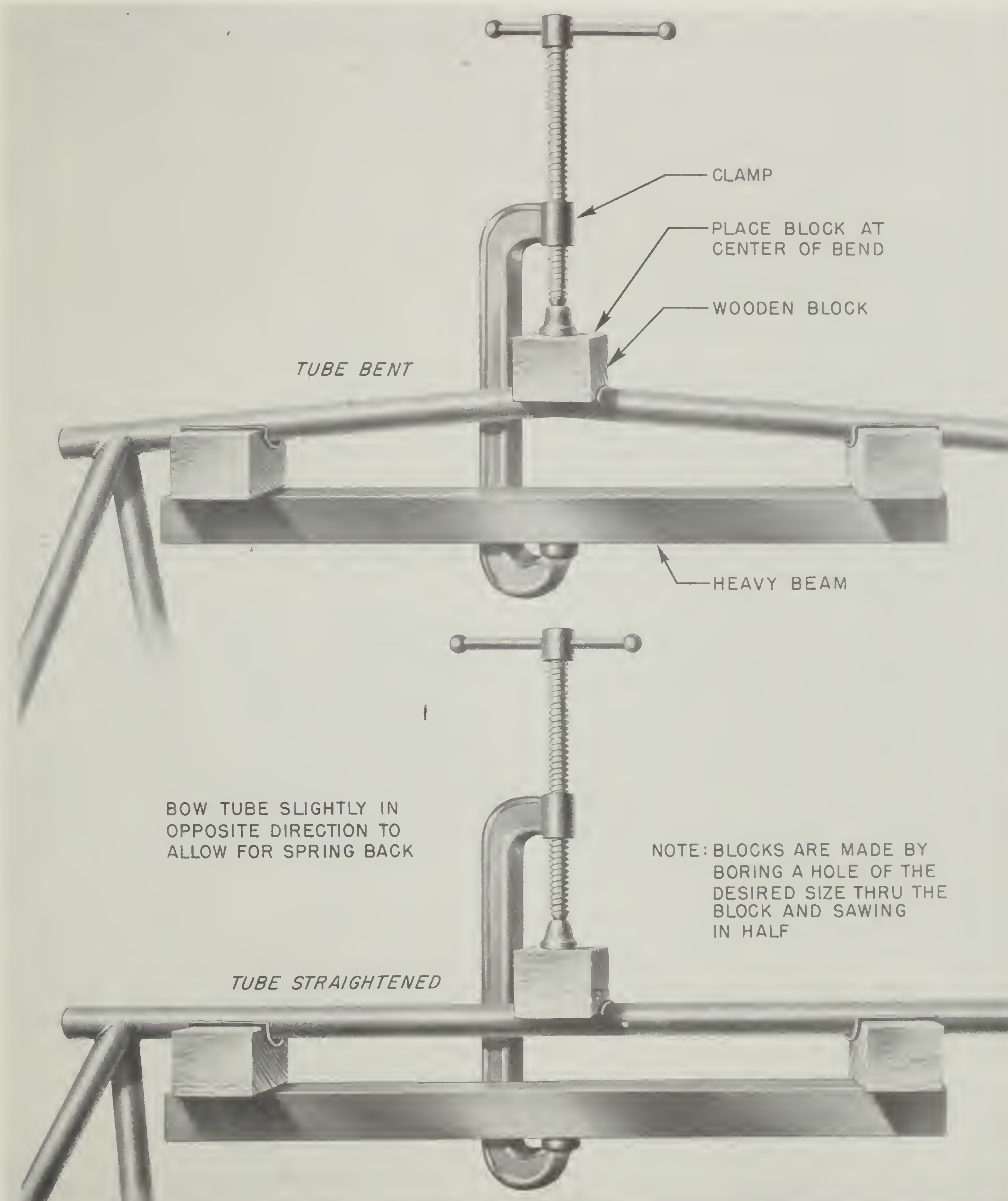


Figure 13—Straightening Bowed Steel Tubes

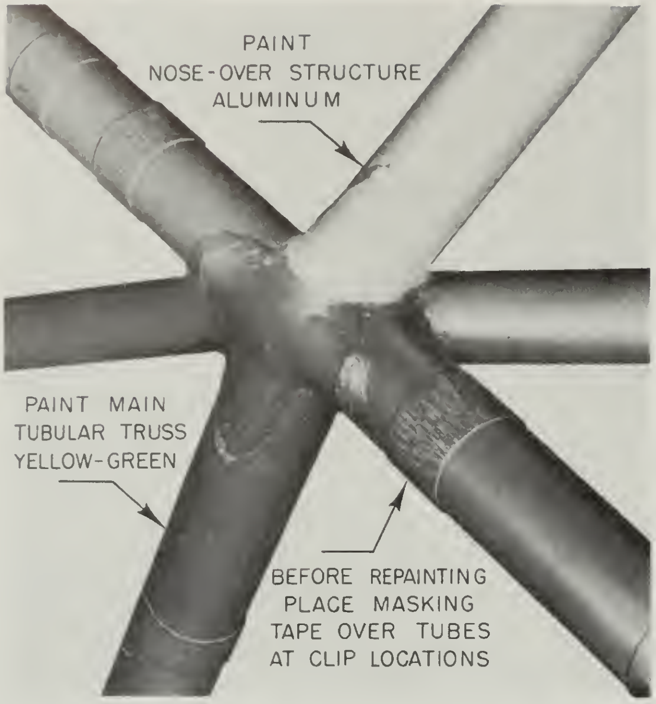


Figure 14—Application of Finish to Steel Tubes

paragraph applicable to cracked tubes. In every case where a bent tube is restored, carefully test all adjacent welded joints for cracks, and repair the cracks as outlined in the following paragraph.

13. SMALL CRACKS AT STEEL TUBING CLUSTER JOINTS.

If it is necessary to check an individual tubing joint for cracks, apply a liberal coat of light oil to the affected area, thoroughly wipe the oil from the joint, and then apply a coat of whiting. A crack in the joint will be made evident by the appearance of oil on the whiting from the crack recess *(see Figure 12)*. If a considerable number of welded joints must be tested, another method of testing, using air pressure, is described in a subsequent paragraph. After locating the crack, remove the whiting and remove all finish from the area by rubbing with steel wool or a wire brush. If the crack is located in an original weld bead, carefully chip, file, or grind out the existing weld bead, and reweld over the crack along the original weld line. When grinding off the original weld bead, take particular care to avoid removing any of the existing tube or gusset material. If the crack is near a cluster joint but away from the original weld bead, remove the finish from the area with steel wool, drill a No. 40 (.098) hole at the ends of the crack, and weld an overlapping doubler over the area as outlined for the repair of dents near a cluster joint. No more than two cracks should be repaired in the same general area. At the conclusion of the repair, apply one coat of zinc chromate primer to the area from which the finish was previously removed. Apply finish coats to match the adjacent surface. These finish coats consist of either two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material *(see Figure 14)*.

14. SHARP DENTS AT A STEEL TUBE CLUSTER JOINT.

Sharp dents at a steel tube cluster joint may be repaired by welding an especially formed X4130 chrome molybdenum steel patch plate over the dented area and surrounding tubes *(see Figure 15)*. To prepare the patch plate, cut a section of X4130 chrome molybdenum steel plate of a thickness equal to or greater than that of the damaged tube. Trim the reinforcing plate as shown *(see Figure 15)* so that the plate extends a minimum of two times the

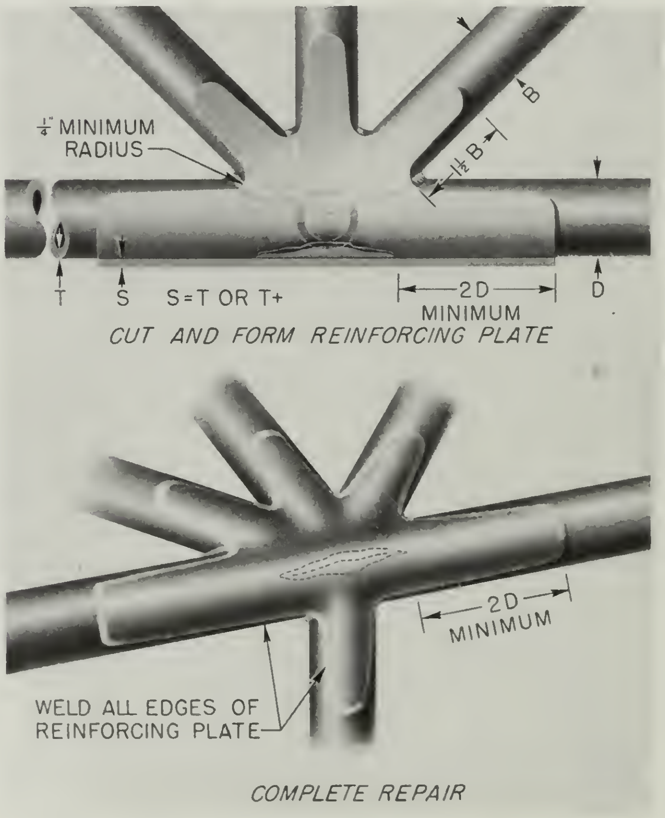


Figure 15—Reinforcing a Dent at a Steel Tube Cluster Joint

diameter of the tube from the nearest edge of the dent, and over adjacent tubes 1-1/2 times the diameter of the tube. On the damaged cluster joint area to be covered by the reinforcing plate, rub off all the existing finish with steel wool. The reinforcing plate may be formed before any welding is attempted, or the plate may be cut and tack welded to one or more of the tubes forming the cluster joint, then heated and pounded around the joint contour as required to produce a smooth contour. Avoid unnecessary heating of the reinforcing plate while forming, but apply sufficient heat and pound the plate so that there is generally a gap of no more than 1/16-inch from the contour of the joint to the reinforcing plate. While forming the plate, exercise care to prevent damage at the apex of the angle formed by any two adjacent "fingers" of the plate. After the reinforcing plate is formed and tack welded to the cluster joint, weld the plate edges to the cluster joint. At the conclusion of the repair, apply one coat of zinc chromate primer to the plate area and to the accessible area from which the finish was previously removed. Apply finish coats to match the adjacent surface. These finish coats consist of either two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material *(see Figure 15)*.

15. SHARP DENTS OR CRACKS IN STEEL TUBE LENGTH.

If a crack appears in a length of steel tube, usually as the result of previously straightening the tube, first drill a No. 40 (.098) hole at the ends of the crack, and then rub the area with steel wool to remove the finish around the tube for a distance of approximately 3 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, first rub the area with steel wool to remove the finish around the tube for a distance of approximately 3 inches on each side of the damage. In order to reinforce the dent or the crack, select a length of a spare X4130 chrome molybdenum steel tube sleeve having an inside diameter approximately equal to the outside diameter of the damaged tube and of the same wall thickness. Diagonally cut the reinforcement steel sleeve at a 30-degree angle on both ends so that the distance of the sleeve from the edge of the crack or dent is not less than 1-1/4 times the diameter of the damaged tube *(see Figure 16)*. Cut through the entire length of the reinforcing sleeve and separate the half sections of the sleeve. Clamp the two sleeve sections to the proper positions on the affected area of the original steel tube. Weld the reinforcing sleeve along the length of the two sides, and weld both ends of the sleeve to the damaged steel tube as shown *(see Figure 16)*. Apply one coat of zinc chromate primer to the reinforcing sleeve area and to the accessible area from which the finish was previously removed. Apply finish coats to match the adjacent surface. These finish coats consist of either two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material.

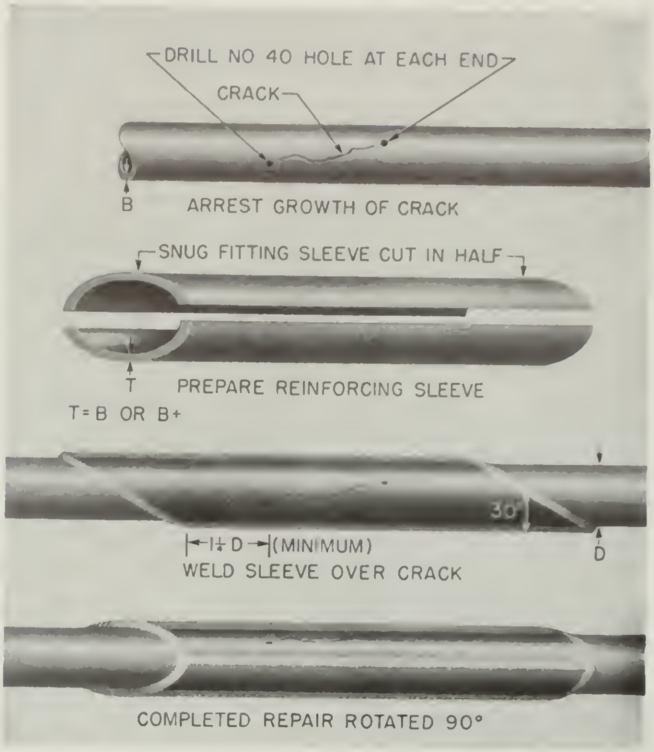


Figure 16—Reinforcing a Dent or Crack in Steel Tube Length

16. SPLICING STEEL TUBES - GENERAL.

Tubular members of the forward fuselage truss may be spliced by partial tube replacement used with internal or external reinforcing steel sleeves, or by the use of an externally telescoping tube replacement of the next larger diameter tubing. Each type of splice has its particular advantage or function, and the methods involved are essentially the same. All splicing is accomplished with electric arc welding or oxyacetylene welding. No bolted joints or splices are permitted. If the original damaged

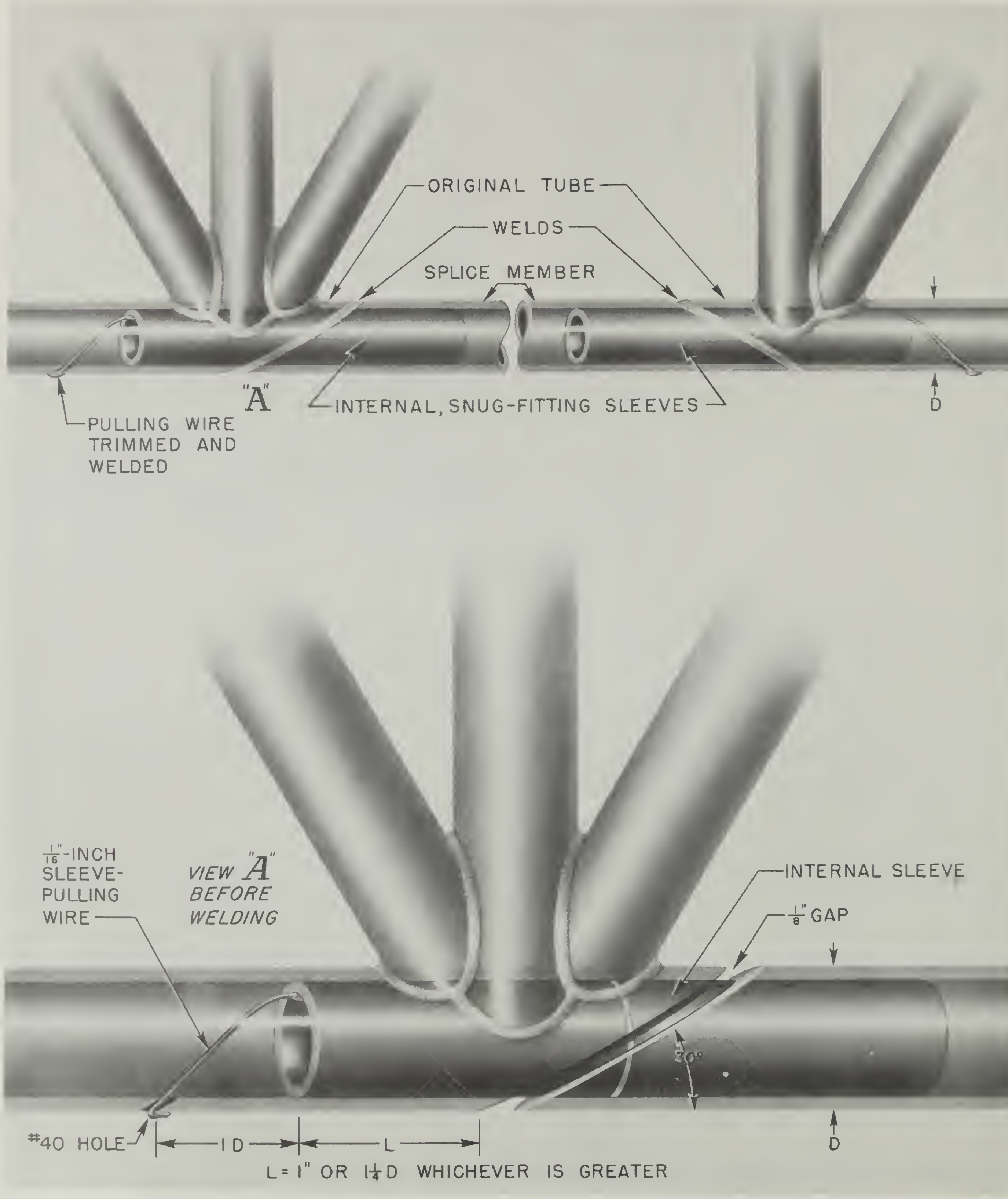


Figure 17—Steel Tube Inner Sleeve Splice

tube accommodates castings or fittings which have been fabricated to fit the tube contour, the spliced replacement tube must be of the same diameter. If no such fittings are applied to the original damaged tube, the externally telescoping splice replacement may be used. Two types of splice welds are permitted, the diagonal and the fishmouth. Being the stronger of the two, the fishmouth splice weld is preferred to the diagonal. However, the nature and location of the damage will determine which type must be used.

Important: Splices must not be made in the middle third of a tube bay, and only one partial replacement tube can be inserted in any one bay of a structural member. If a member is damaged at a joint so that it is impossible to retain a stub to which another member can be attached, replace the tube if it is a web member; and if the tube is a continuous longeron, locate the splice in the adjacent bay. Make no splice by butt welding any member between station points. Make no repairs to the engine mount members located within 3 inches of the mounting ring, and make no repairs to the mounting ring. Misalignment of a tubular structure due to contraction and expansion of the metal during welding can be prevented by using wood braces with notches in the ends to hold the tubes in position. When new tubes are used to replace bent or damaged tubes, the original alignment of the structure must be maintained and checked. This can be done by measuring the distance between points of corresponding members on an undamaged aircraft.

17. SPLICING STEEL TUBE BY INNER SLEEVE METHOD.

If the damage to a steel tube is such that partial replacement of the tube is necessary, the inner sleeve splice is recommended especially where a smooth tube surface is desired *(see Figure 17)*. Diagonally cut out the damaged portion of the tube with a hack saw, locating the cuts away from the middle third of the affected tube bay. Burr and clean the edges of the cuts. Diagonally cut a replacement X4130 chrome molybdenum steel tube to match the diameter wall thickness and length of the removed portion of the damaged tube. At each end of the replacement tube, allow a 1/8-inch gap from the diagonal cuts to the stubs of the original tube. Select a length of X4130 chrome molybdenum steel tubing of the same wall thickness and of an outside diameter approximately equal to the inside diameter of the damaged tube. This inner sleeve tube material should fit snugly within the original tube, with a maximum tolerance of 1/64-inch. From this inner sleeve tube material, cut two sections of tubing, each of such a length that the ends of the inner sleeve will be a minimum distance of 1-1/4 tube diameters from the nearest end of the diagonal cut *(see Figure 17)*. Dip the replacement tube and the inner sleeves into hot (74°C, 165°F) raw linseed oil; then wipe the oil from the outer circumference of the tubes. Make a mark on the outside of the diagonally cut original tube stub midway along the diagonal cut *(see Figure 18)*. At a minimum distance of 2-1/4 times the tube diameter measured from the nearest end of the diagonal cut, center punch the tube, and start drilling the No. 40 hole at a 90-degree angle. After a shallow hole is started from which the drill will not jump out, slant the drill toward the cut and drill at a 30-degree angle. Slanting the hole in this manner aligns the edges of the hole with the line of pull of the sleeve-pulling wire, and prevents the wire from scraping the hole edges. Burr the edges of the hole with a round, needle-point file. Obtain a length of 1/16-inch welding or brazing wire, insert one end into the drilled hole, and push the wire out the end of the tube *(see Figure 18)*. Weld the end of the wire to the inner side of the reinforcing sleeve. Chamfer the end of the sleeve as an aid in sliding the tube into position. With thin paint, metal dye, or emery paper, make a narrow mark around the center of the reinforcing sleeve. Slip the sleeve into the replacement tube so that the welded wire is 180 degrees from the drilled hole. If the inner sleeve fits very tightly in the replacement tube, chill the sleeve in cold water and heat the replacement tube and the stubs of existing tube. The resultant expansion of the tube stubs and the contraction of the inner sleeve temporarily allow more clearance from the inner sleeve to the inside wall of the tube stubs. Align the original tube stubs with the replacement tube. Pull on the exposed end of the sleeve-pulling wire until the center mark on the sleeve is directly in line with the center mark on the diagonal cut *(see Figure 18)*. When this occurs, the inner sleeve is centered beneath the joint. Sharply bend the pulling wire over to hold the sleeve in position. At each side of the replacement tube, weld the inner sleeve to the tube stubs through the 1/8-inch gap between the stubs *(see Figure 18)*. Completely fill the 1/8-inch gap and form a weld bead over gap. After the joint is welded, snip off the pulling wire flush with the surface of the tube and weld over the hole. To the outside of the replacement tube, apply one coat of zinc chromate primer, and match adjacent finish with two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material.

18. SPLICING STEEL TUBE BY OUTER SLEEVE METHOD.

If partial replacement of a tube is necessary, an outer sleeve splice may be used, where conditions warrant, as an alternate splice to the inner sleeve splice and the splice using a larger diameter replacement tube. However, the outer sleeve splice requires the most amount of welding; and therefore, it should be used only where the other splicing methods are not suitable. Squarely cut out the damaged section of the tube, locating the cuts away from the middle third of the tube bay. Cut a replacement X4130 chrome molybdenum steel tube section to match the outside diameter, wall thickness, and length of the removed tube. This replacement tube must bear against the stubs of the original tube with a total tolerance not to exceed 1/32-inch. Dip the replacement tube into hot (74°C, 165°F) raw linseed oil; then wipe the oil from the outer circumference of the tube. Select a length of X4130 chrome molybdenum steel tubing of an inside diameter approximately equal to the outside diameter of the damaged tube and of the same wall thickness. This outer sleeve tube material should fit snugly about the original tube with a maximum tolerance of 1/64-inch. From this outer sleeve tube material, diagonally or fishmouth cut two sections of tubing, each of such a length that the nearest ends of the outer sleeve are a minimum distance of 1-1/4 tube diameters from the ends of the cut on the original tube *(see Figure 19)*. Use a fishmouth-cut sleeve wherever possible. Burr all the edges of the sleeves, replacement tube, and original tube stubs. Slip the two sleeves over the replacement tube, line up the replacement tube with the original tube stubs, and slip the sleeves out over the center of each joint *(see Figure 19)*. Adjust the sleeves to suit the area and to provide maximum reinforcement. Do not tack weld the two sleeves to the replacement tube before welding. Apply a uniform weld around both ends of one of the reinforcing sleeves and allow the weld to cool. Then weld around both ends of the remaining reinforcing tube *(see Figure 19)*. Allowing one sleeve weld to cool before welding the remaining tube prevents undue warping. To the outside of the replacement tube, apply one coat of zinc chromate primer and match adjacent finish with two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material.

19. SPLICING STEEL TUBE USING LARGER DIAMETER REPLACEMENT TUBE.

This method of splicing steel tubes requires the least amount of cutting and welding. However, this splicing method cannot be used where the damaged tube is cut too near the adjacent cluster joints or where bracket mounting provisions make it necessary to maintain the same replacement tube diameter as the original. As an aid in installing the replacement tube, squarely cut the original damaged tube, leaving a minimum short stub equal to 2-1/2 tube diameters on one end and a minimum long stub equal to 4-1/2 tube diameters on the other end *(see Figure 20)*. The cuts must be away from the middle third of the affected tube. Select a spare length of X4130 chrome molybdenum steel tubing having an inside diameter approximately equal to the outside diameter of the damaged tube and of the same wall thickness as or greater than the damaged tube. This replacement tube material should fit snugly about the original tube with a maximum tolerance of 1/64-inch. From this replacement tube material, diagonally or fishmouth cut a section of tubing of such a length that each end of the tube is a minimum distance of 1-1/4 tube diameters from the end of the cut on the original tube. Use a fishmouth-cut replacement tube wherever possible *(see Figure 20)*. However, a diagonally cut tube may also be used *(see Figure 21)*. Burr the edges of the replacement tube and the original tube stubs. If a fishmouth cut is used, file out the sharp radius of the cut with a small round file. Dip the replacement tube into hot (74°C, 165°F) raw linseed oil; then wipe the oil from the outer circumference of the tube. Spring the long stub of the original tube from the normal position; slip the replacement tube over the long stub, then back over the short stub. Center the replacement tube between the stubs of the original tube. In several places tack weld one end of the replacement tube; then weld completely around the end. In order to prevent distortion, allow the weld to cool completely; then weld the remaining end of the replacement tube to the original tube *(see Figure 20)*. To the outside of the replacement tube, apply one coat of zinc chromate primer and match adjacent finish with two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material.

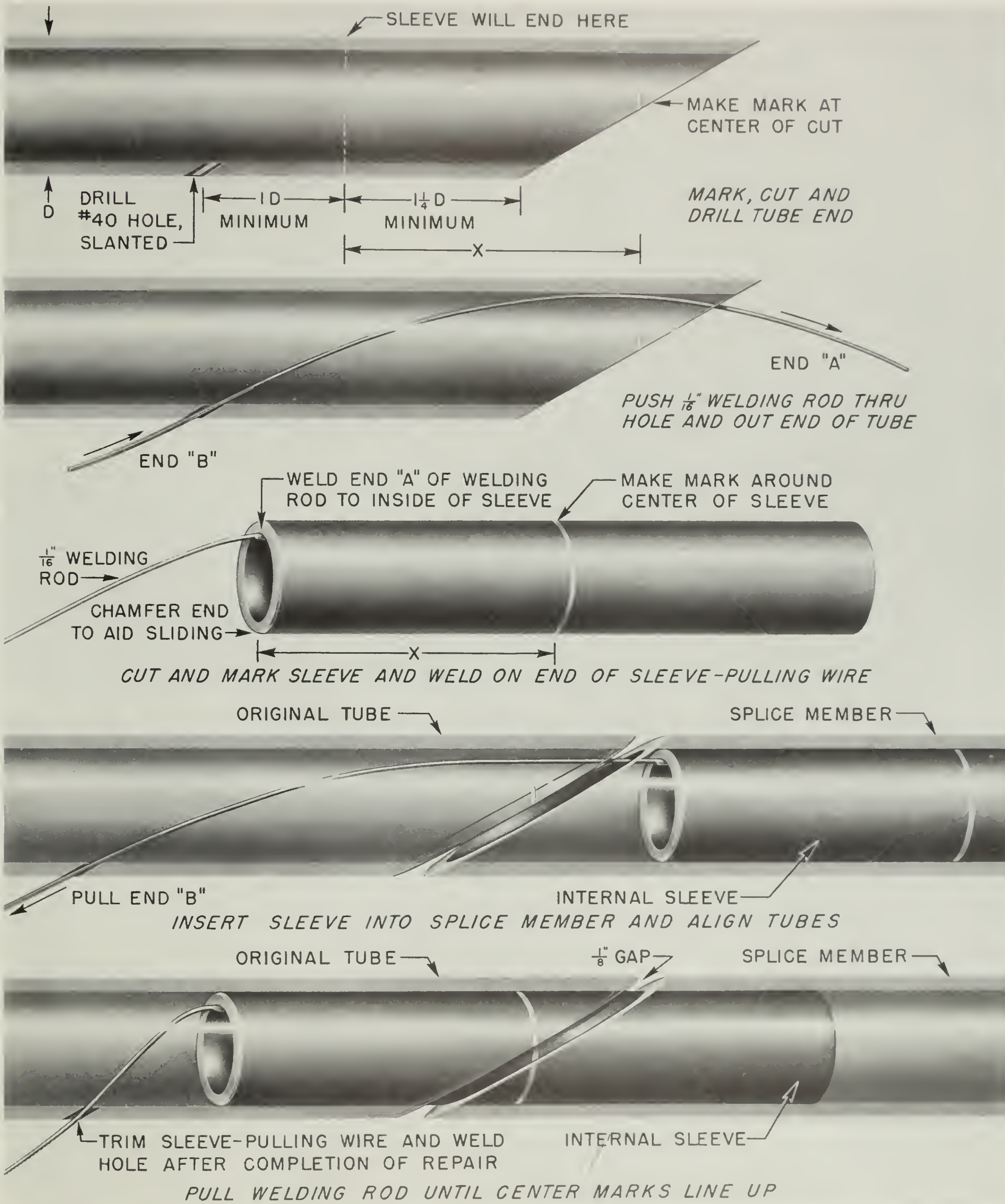


Figure 18—Centering Inner Sleeve in Steel Tube

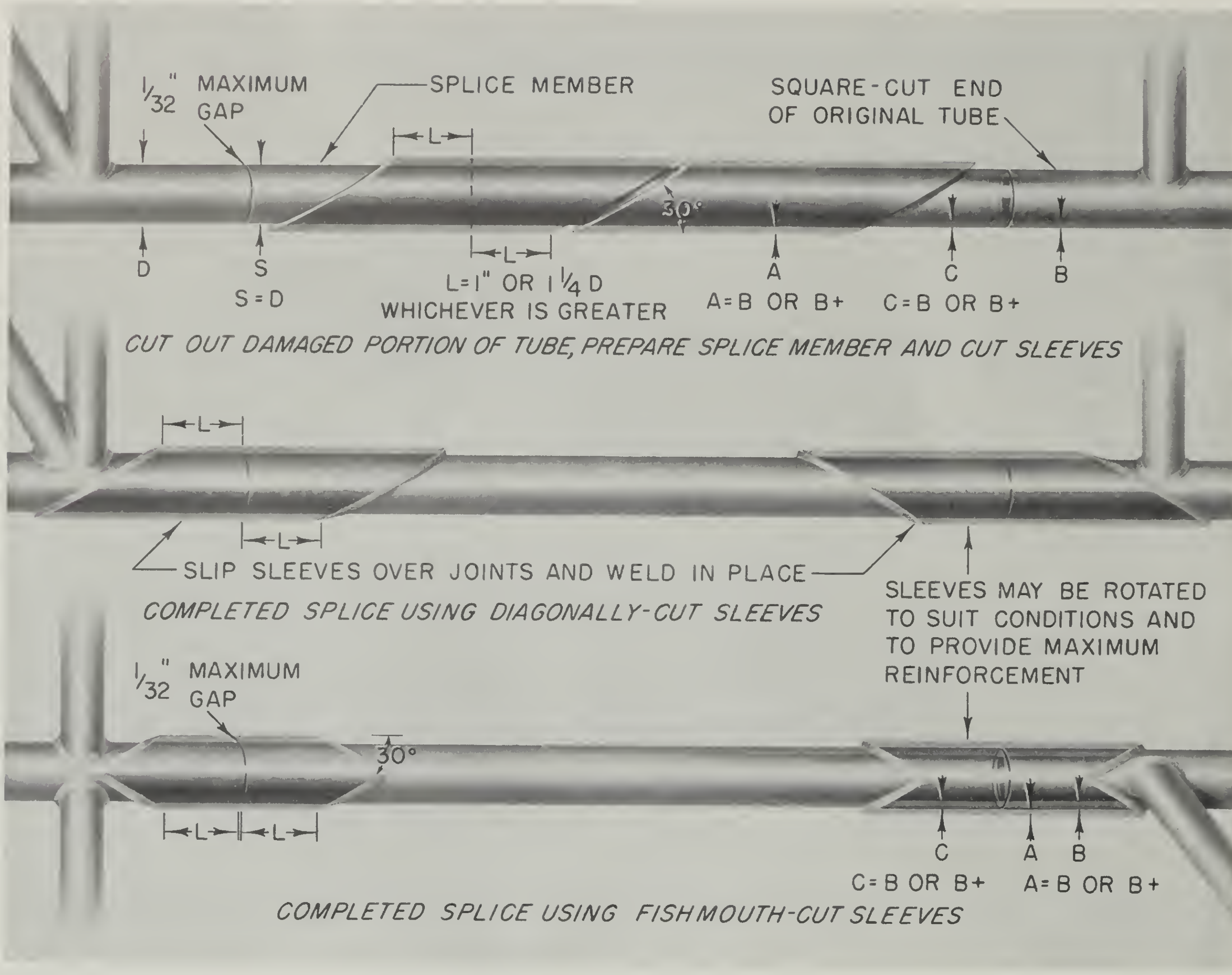


Figure 19—Steel Tube Outer Sleeve Splice

20. REPLACING STEEL TUBES.

Tubes damaged beyond the limitations specified in the preceding paragraphs on tube splicing should be replaced. Tube replacement is necessary where an original tube stub is too short to attach a replacement and where splice welds will be made in the middle third of a member. When it is necessary to remove a member at a joint or from a cluster of tubes, use a sharp fine-toothed hack saw and remove the tube carefully and completely from the structure. While cutting out the tube, exercise caution to prevent any damage to adjacent tubes or welds. Where new welds are to be made over the location of existing welds upon the insertion of the new member, completely chip or file off the old welds. Dip the replacement tube into hot (74°C, 165°F) raw linseed oil; then wipe the oil from the outer circumference of the tube. When installing a new tube member, allow a clearance of 1/32-inch at either end for expansion. Unless a welding jig is available, the actual process of welding should be accomplished in as systematic a manner as possible pertinent to the application of heat and the resultant possible distortion. After the new tube has been welded in place, clean the welded joints with a wire brush. To the outside surface of the replacement tube, apply one coat of zinc chromate primer, and match adjacent finish with two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material (see Figure 14). Check the alignment of the truss as outlined in a subsequent paragraph.

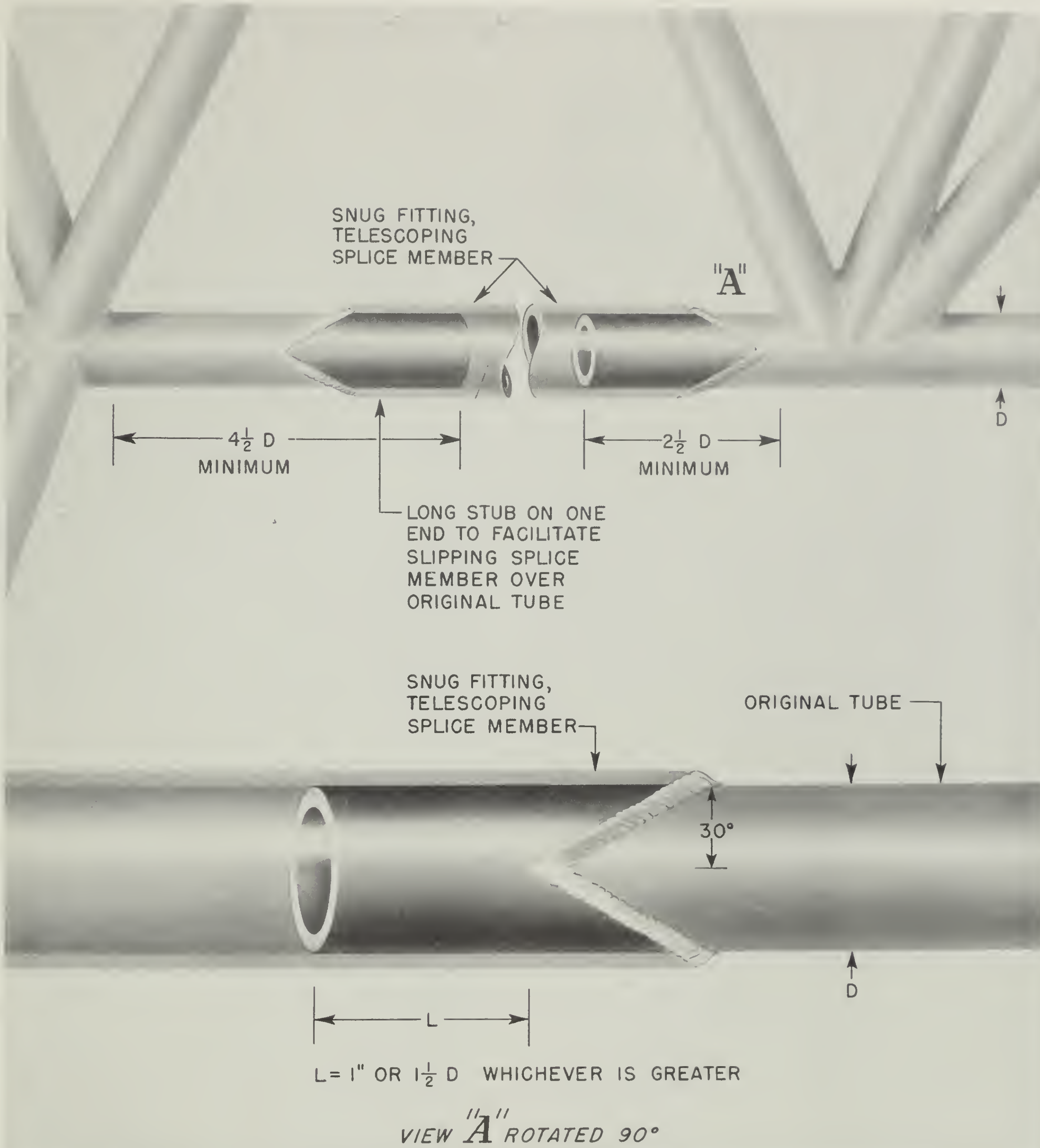


Figure 20—Steel Tube Fishmouth Splice Using Larger Diameter Replacement Tube

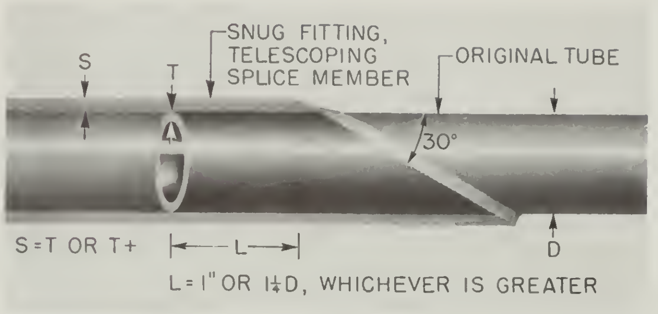


Figure 21—Steel Tube Diagonal Splice Using Larger Diameter Replacement Tube

21. TESTING WELDED STEEL JOINTS.

In the event of an accident not visually indicating damage, the fuselage frame should be tested for the security of the welded joints as outlined in the previous paragraph on small cracks at steel tubing cluster joints. Another method of testing is as follows: Remove one of the self-tapping screws provided at the extremities of the main members, and apply an air pressure of 30 lbs./sq.in. to the interior of the tubular framework. Thoroughly mix one cup of liquid castile soap with one pint of water and apply this solution with a soft-bristled brush to all welded joints or to the joints in question. A cracked weld will become evident by the formation of large soap bubbles. Mark the weld for subsequent repair. The entire forward fuselage truss should be airtight to prevent acceleration of corrosion. Upon completion of testing, reinsert any self-tapping screws that were removed. Typical welding flaws which can be easily located by this method are illustrated *(see Figure 11)*.

22. PROTECTING STEEL TUBE AGAINST CORROSION.

Prior to welding, dip all replacement tubes and inner reinforcing sleeves into hot (74°C, 165°F) raw linseed oil; then wipe the oil from the outer circumference of the tubes. This treatment protects the interior of the tubes from corrosion. Outer reinforcing sleeves do not require this treatment. After welding, apply one coat of zinc chromate primer to all outside unfinished surfaces, and match adjacent finish with two coats of yellow-green lacquer or two coats of clear lacquer containing a minimum of 4 ounces of aluminum paste per gallon of spraying material *(see Figure 14)*.

23. ALIGNMENT OF ENGINE MOUNT AND FRONT FUSELAGE TRUSS.

In the event that structural damage necessitating major repair occurs to the forward fuselage truss, care must be exercised to maintain proper alignment. In cases of major misalignment concluding repair, replace the structural members affected. Minor misalignments which cannot be detected by a visual inspection can usually be detected by means of simple measuring devices and straight edges. One practical method is to measure a corresponding member on undamaged aircraft and compare measurements. If necessary, the application of heat may be utilized as a means of relieving minor misalignment of a tube. Apply heat to a short distance on either side of the outside point of maximum curvature. Heat to a dull red and remove the torch. Allow the tube to air-cool. DO NOT QUENCH. Contraction of the metal will force the tube into proper alignment. The application of heat may be accomplished without disassembling the aircraft provided all light metal parts or items liable to damage by proximity to heat are removed or otherwise protected by asbestos sheeting or damp rags.

24. EXHAUST MANIFOLD ASSEMBLY CONSTRUCTION.

The exhaust manifold and collector ring assembly constitutes one part of the engine installation, and collects and carries away the exhaust gases from the engine. The entire assembly is constructed of stainless steel and is of the slip joint type *(see Figures 22 and 23)*. The collector is made from sheet stock, and the small component parts are made from bar stock and

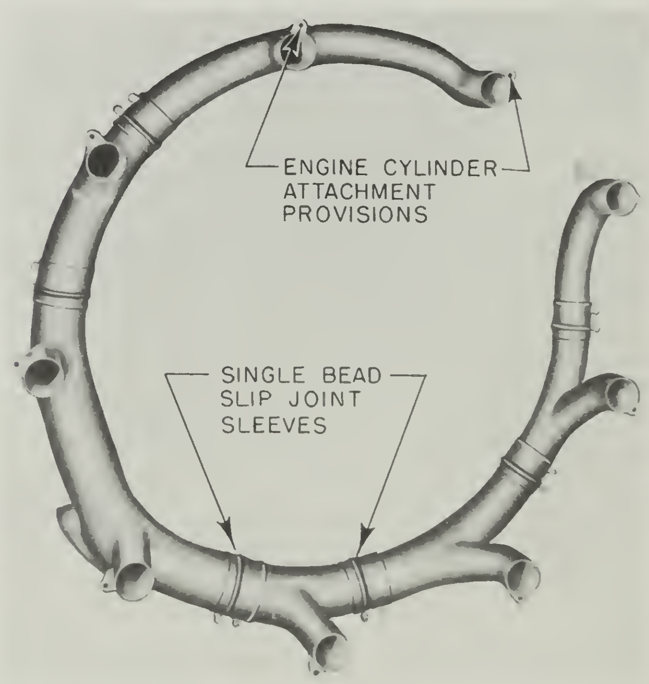


Figure 22—Exhaust Manifold Assembly

forgings. The collector is assembled by means of arc and gas welding. Arc welding is used on all fittings such as exhaust port bolt flanges, collar bolt tubes, doublers, and lap joints. Gas welding is employed on all butt joints of the main collector ring assembly.

25. EXHAUST MANIFOLD REPAIR.

All repairs to the exhaust manifold assembly other than part replacement are accomplished by welding. The type of welding used is not critical, as both gas and arc welding are satisfactory. However, welded repairs should follow the design of the original construction as far as possible. Permissible welded repairs consist of welding cracks around exhaust port nipples, welding reinforcement or doublers over fatigue points, welding patches over bullet or other small holes, and repairing cracks in the main body of the collector ring. For gas welding the manifolds, use a mixture of Cromaloy (Linde Air Products) or equivalent, and Inconel flux with sodium silicate and water or with shellac as a binder. Use a neutral flame to prevent oxidation and carbon pickup. Point the flame in the direction of the weld progress to preheat the metal. The weld should be made as rapidly as possible to prevent an excessive puddling and carbon pickup. For arc welding the exhaust manifold, use a coated, stainless steel electrode of 1/16- or 3/32-inch diameter. After welding has been accomplished, the parts must be passivated to improve their corrosion resistance. Immerse the parts in a 17 to 20 percent aqueous solution of nitric acid at 49° to 66°C (120° to 150°F) for 21 minutes, and thoroughly rinse in water. The use of hot water for rinsing gives better results and promotes rapid drying.

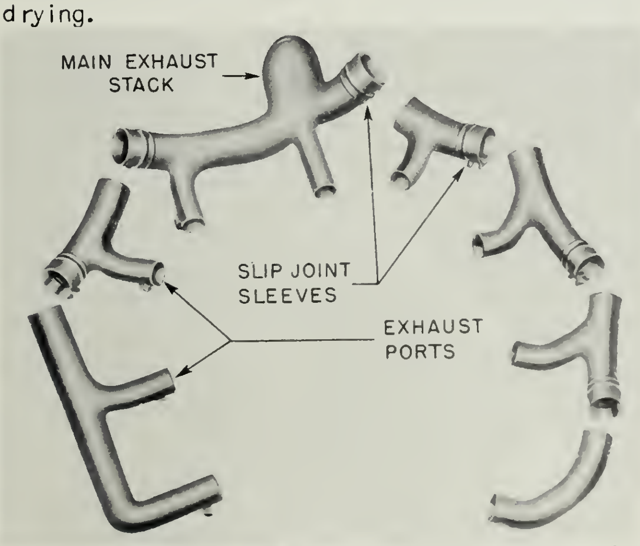


Figure 23—Exhaust Manifold Assembly Exploded

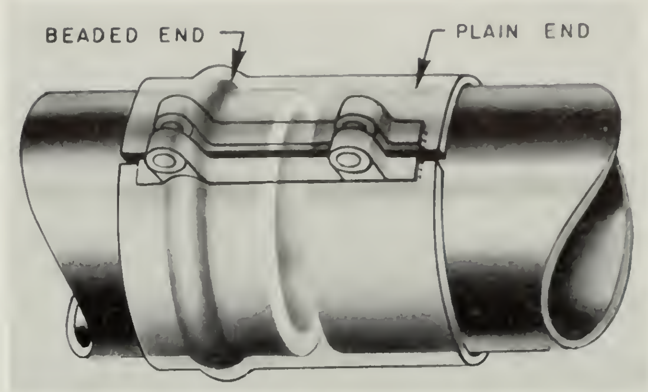


Figure 24—Exhaust Manifold Slip Joint

26. EXHAUST MANIFOLD COLLAR REPLACEMENTS.

The collar assemblies which join the various sections of the collector ring are of the single-beaded type which allows for slippage at all joints *(see Figures 23 and 24)*. Worn collars should be replaced and new bolts used to secure them. The collar bolt tubes may be replaced if they break. Grind the original weld from the collar before welding the new bolt tube into place. The collar may be used as a jig by inserting the bolts and by tacking the new bolt tube into position. The bolts may then be removed and the weld completed.

27. EXHAUST MANIFOLD REASSEMBLY.

Before reassembly of repaired members, all welded repairs should be lightly sandblasted to clean them. After sandblasting, the parts must be passivated as outlined in a preceding paragraph. Upon reinstallation, care should be taken to align all parts properly to prevent excessive wear. The brass nuts securing the exhaust port flanges should be tightened to 165-175 inch-pounds. Brass nuts should be used on all collar attaching bolts and should be tightened one castellation beyond finger tightness.

28. ENGINE FIREWALL CONSTRUCTION.

A corrosion-resistant, .019 inch thick sheet steel firewall is located at the forward end of the main fuselage truss assembly at fuselage station 0. This firewall forms a fireproof bulkhead between the engine and the pilot's compartment. Five stiffeners are located on the aft side of the firewall to add rigidity and strength and to keep cans from forming in the bulkhead between attachment points *(see Figure 25)*. Four round holes in the firewall permit the firewall to be slipped over the four

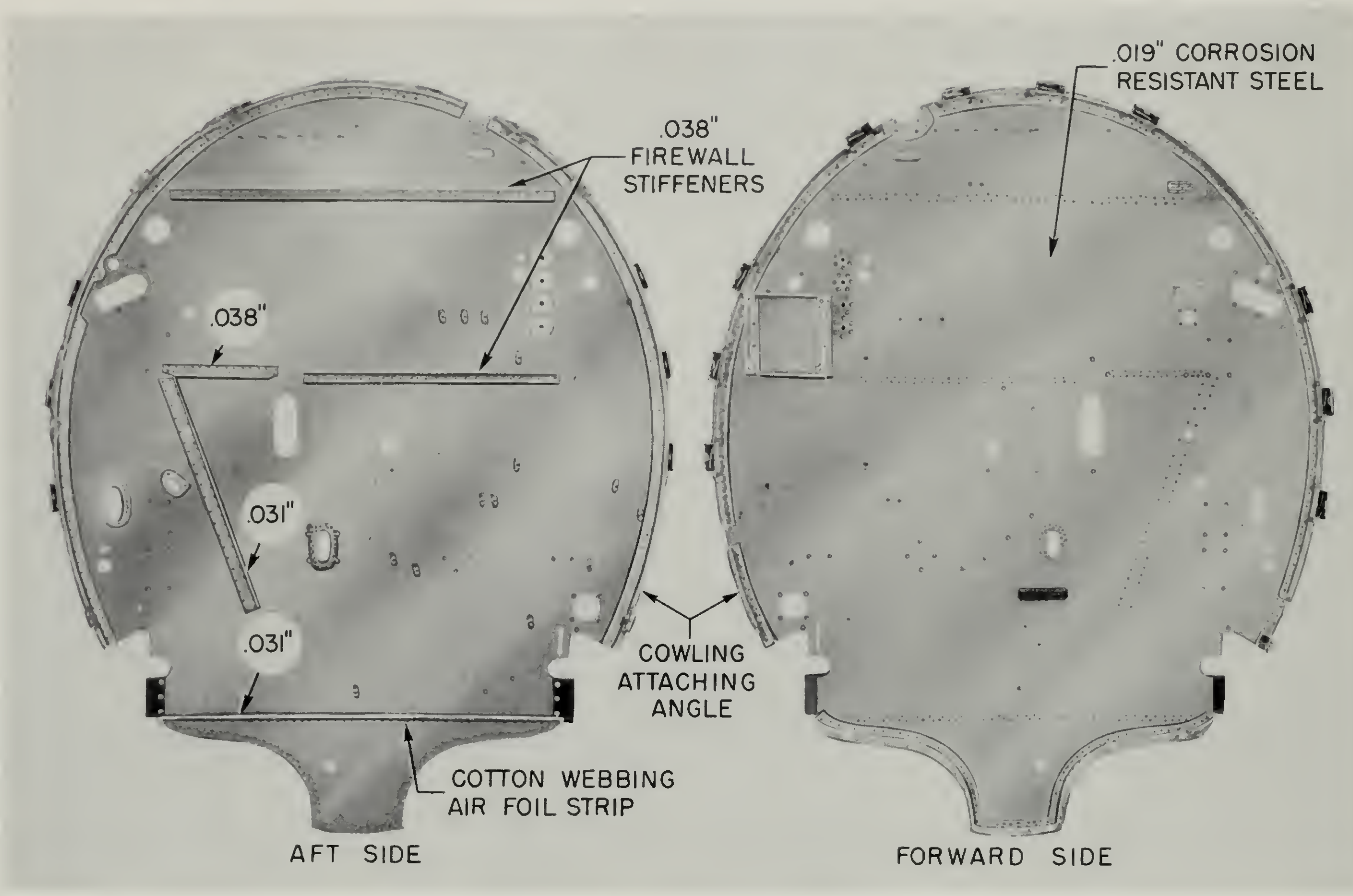


Figure 25—Engine Firewall Assembly

engine mount attachment bolt fittings. Four U-bolt assemblies attach the firewall to the foremost diagonal tube member of the forward fuselage truss. A break in the horizontal firewall stiffening angles on the aft side of the firewall permits the firewall to be clamped flat against the surface of this tube. A corrosion-resistant attaching angle is spot-welded around the periphery of the firewall. Dzus fastener attachment springs are located at intervals around this angle to provide for the attachment of the engine accessory compartment cowling, the landing wheel fairings, and the fairing located around the periphery of the fuselage just aft of the firewall *(see Figure 25)*.

29. FIREWALL WEB REPAIR.

Removal of the firewall cannot readily be accomplished to permit repair members to be spot-welded to the firewall. Repair members must, therefore, be secured to the firewall by means of AN441-4 iron rivets, 1/8-inch diameter. If these rivets are not available, Type AN442-AD4 (A17S aluminum alloy) rivets may be substituted if they are dipped in zinc chromate primer prior to driving. This rivet corrosion protection is necessary because of the contact of dissimiliar metals. Before drilling holes in the firewall, check the opposite side for clearance on all engine accessories, electrical junction boxes, and other equipment. Holes through the firewall may be repaired by riveting on a flat patch of .019 inch thick or thicker corrosion-resistant steel *(see Figure 26)*. Cut away the damage to form a circular opening. A slow-speed hole saw or a round file may be used to accomplish this. Cut a patch of .019 inch thick or thicker corrosion-resistant steel sheet. Allow a 3/4-inch overlap around the edge of the hole; and with a No. 30 drill, drill a row of holes around the patch in the lap area. Space the holes approximately 5/8-inch apart and 3/8-inch in from the edge of the patch to provide proper edge distance for the rivets *(see Figure 26)*. Upon completion of drilling, remove the patch and clean all surfaces of chips. Realign the patch and insert two or more skin fasteners to hold the patch in place until two or more rivets have been driven. Insert and upset an AN441-4 iron rivet, 1/8-inch diameter, in each hole. The rivets may be driven from

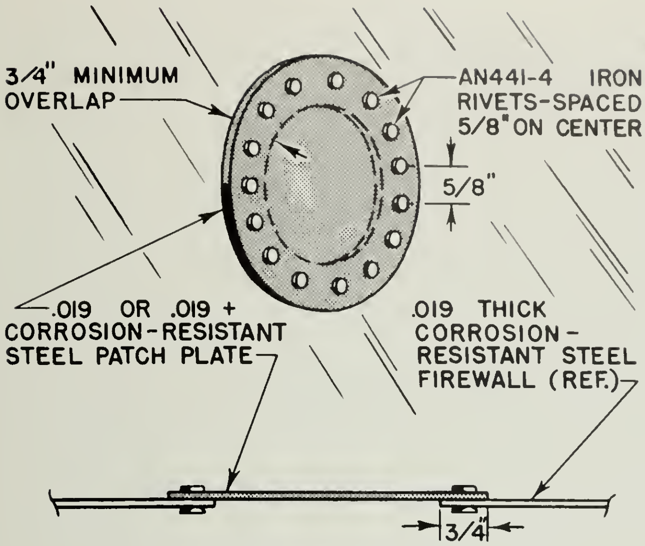


Figure 26—Firewall Web Repair

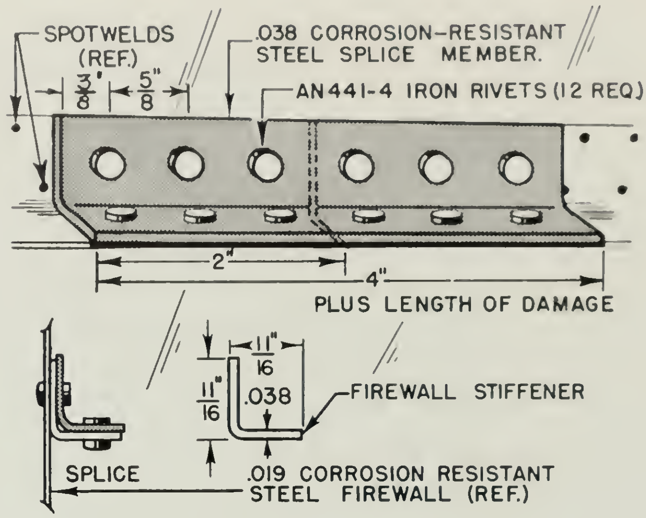


Figure 27—Firewall Wide-flanged Stiffener Splice

either side of the firewall, depending upon working area available.

30. FIREWALL STIFFENER REPAIR.

Inasmuch as all firewall stiffening angles are spot-welded to the firewall, damaged angle stiffeners cannot be removed without subsequent damage to the firewall. To splice these stiffeners, proceed as follows *(see Figure 27)*: Smooth out the damage with a file and remove all sharp edges. Fabricate a similar stiffener of corrosion-resistant steel of the same gage and cross-sectional area as that of the damaged member. Center this splice member over the damage. The length of the splice member should be such that it will extend at least 3 inches on each side of the damage and should be secured to the damaged stiffener by three equally spaced rivets on the upstanding flange on each side of the damage and attached to the firewall flange by three equally spaced rivets on each side of the damage *(see Figure 27)*. If the upstanding flange of the stiffener is not at least 5/8-inch in height, the rivets in the upstanding flange must necessarily be omitted. Where this condition exists, the splice member should be thicker than the damaged angle section, one additional rivet must be used on each side of the damage, and the length of the splice member increased accordingly *(see Figure 28)*. When drilling corrosion-resistant steel, use a slow-speed drill to avoid burning the tip.

31. COWLING ATTACHING FIREWALL ANGLE REPAIR.

Repairs to the corrosion-resistant steel cowling attaching angle which is spot-welded around the periphery of the firewall are made in the same manner as set forth in the preceding paragraph on firewall stiffeners. The splice member must be curved slightly to conform to the curve of the attaching angle. This can be accomplished with the use of a metal shrinker.

32. REPAIR MATERIALS FOR STEEL STRUCTURES.

The following list is a summation of the materials required for the repair of the engine mount, the front fuselage truss, the exhaust manifold assembly, and the engine firewall.

MATERIALS	REMARKS
SHEET, STAINLESS STEEL	NAA SPEC. SS 2118
TUBE, CHROME MOLYBDENUM STEEL X4130	DIAMETER AND WALL THICKNESS TO SUIT (SPEC. 57-180-2)
RIVETS, AN441-4, 1/8-INCH DIAMETER	MILD STEEL, LENGTH AS REQUIRED
PRIMER, ZINC CHROMATE	USE AS DIRECTED (SPEC. 14080)
WHITING	
ROD, WELDING	STAINLESS STEEL-COATED, AND CHROME ALLOY
SHEET, CHROME MOLYBDENUM STEEL X4130	SPEC. AN-QQ-S-685
FLUX, CROMALOY (LINDE AIR PRODUCTS CO., NEW YORK)	INCONEL WITH SODIUM SILICATE AND WATER, OR WITH SHELLAC AS A BINDER
OIL, LINSEED	SPEC. JJJ-O-336
LACQUER, CLEAR	SPEC. 3-158
PASTE, ALUMINUM	SPEC. TT-A-466; TYPE B
LACQUER, YELLOW-GREEN	SPEC. 3-100-H

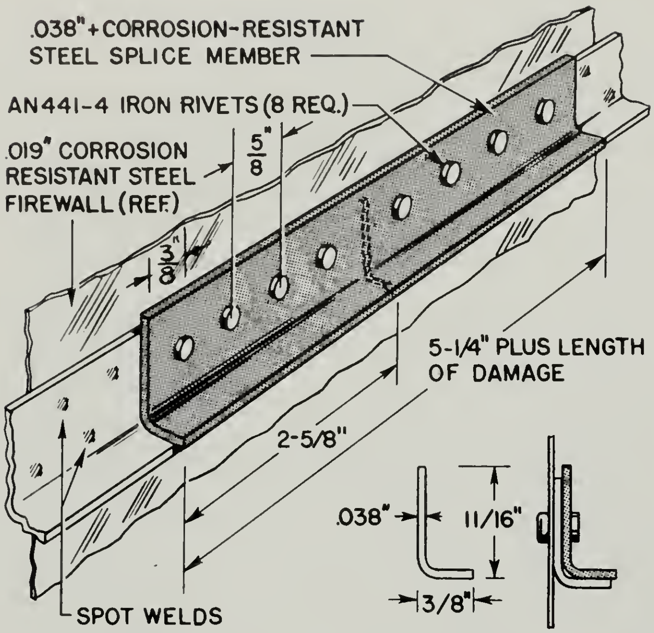


Figure 28—Firewall Narrow-flanged Stiffener Splice

33. REPAIR TOOLS FOR STEEL STRUCTURES.

To repair the various members of the engine mount properly, the forward fuselage truss, the exhaust manifold assembly, and the engine firewall, all or part of the following tools are required.

TOOL	REMARKS
EQUIPMENT, GAS AND ARC WELDING	SEE FIGURES 6 AND 10
TOOLS, SHEET METAL CUTTING, FORMING, AND BENDING AS REQUIRED	SEE SECTION 1
TOOLS, RIVETING, STANDARD	SEE SECTION 1
BRUSH, WIRE	
BRUSH, PAINT	
STEEL WOOL	

34. ENGINE COWLING CONSTRUCTION.

A streamlined, spot-welded, and flush-riveted, removable cowling assembly encloses and streamlines the engine, the exhaust collector ring, and the engine accessory compartment *(see Figure 30)*. All removable parts of the cowling are secured by means of flush-type dzus fasteners and interior and exterior clamping bolts. Most of the cowling sections are made from .032 and .040 inch thick 24ST alclad. All parts which are subjected to heat from the exhaust stack or from the fixed fuselage gun blast are made of .025 and .038 inch thick corrosion-resistant steel. Cowling arrangement, material, material thickness, and major assembly part numbers are illustrated *(see Figures 29 and 30)*. Padded support brackets on the engine support the engine cowling assembly, which is held in position by aligning pins and is securely clamped around the periphery of the engine by one exterior and two interior clamping bolts *(see Figure 29)*. The flush-type dzus fasteners which secure the engine accessory compartment cowling

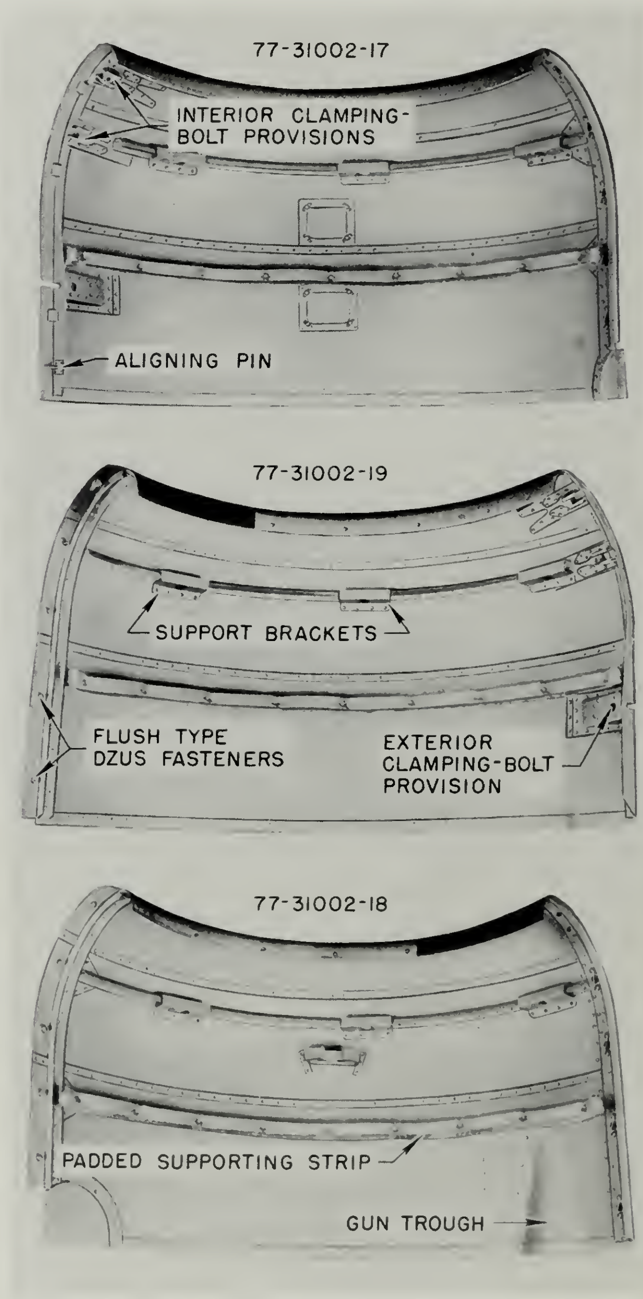


Figure 29—Engine Cowling Construction

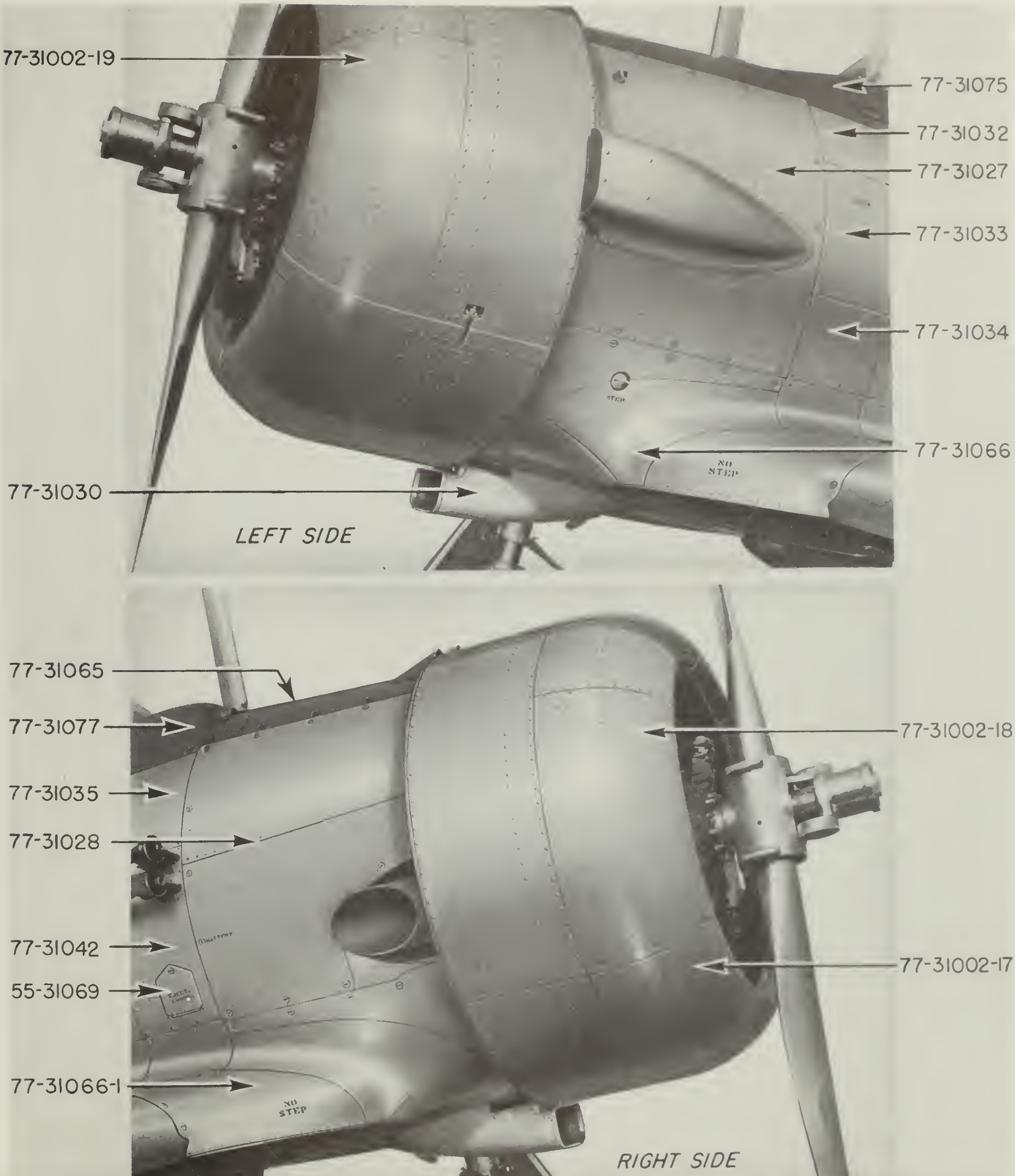


Figure 30—Engine Cowling Assembly

are secured to four stainless steel channels which extend from the firewall to the baffle plate ring just aft of the exhaust collector ring.

35. ENGINE COWLING REPAIR.

Permissible repairs consist of re-forming dented areas, replacements of damaged or broken dzus fasteners, and the patching of holes. Inasmuch as all parts of the cowling assembly are in the high-pressure area of the air stream around the engine, all patches to the cowling must be of the flush type, using a filler piece to replace the removed material and an overlapping piece on the inside of the cowling. Refer to Section III for flush-type skin repairs and to Section IX for removal and replacement of damaged or broken cowling dzus fasteners. Hole patches to corrosion-resistant steel sections of the cowling may be secured by spot welding if such equipment is available, or a single row of Type AN450D8 countersunk steel rivets may be used around the damage to secure the steel patch plate. Some of the corrosion-resistant steel gun blast plates are doubled over the 24ST material of the engine cowling. If these sections become damaged, it will be necessary to remove all attaching rivets and replace the blast plate with a new part.

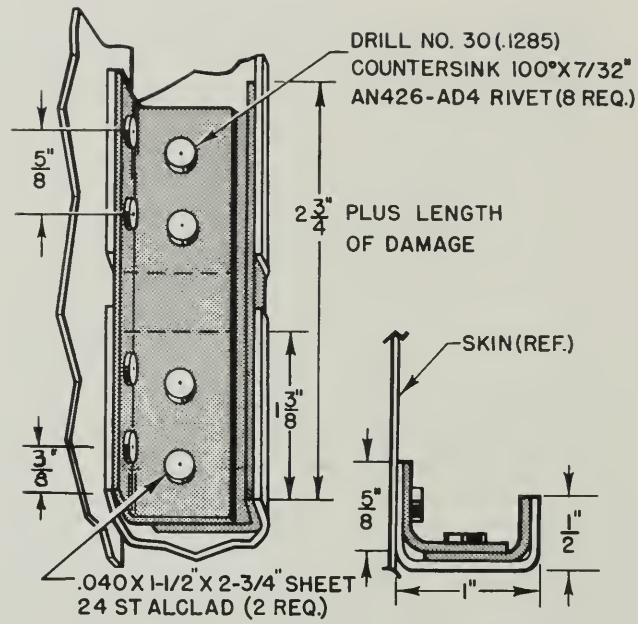
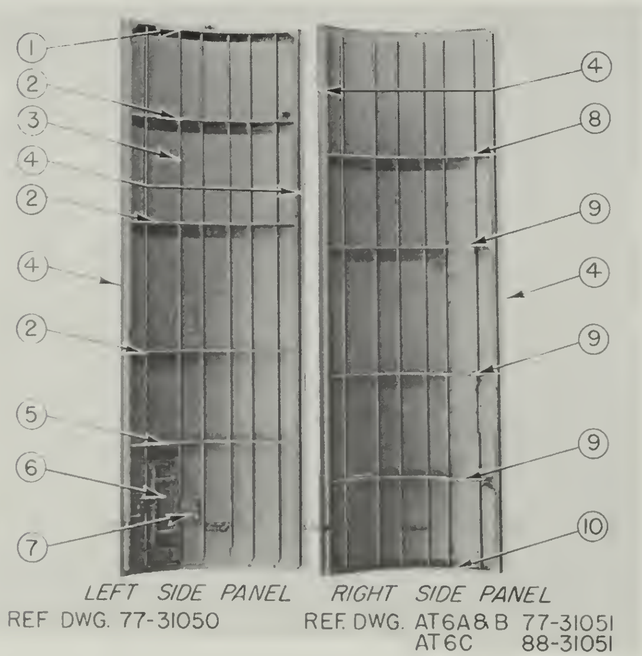


Figure 32—Front Fuselage Side Panel Forward Former Splice



1. 66-31073 - FORMER
2. 52-31200-4 - FORMER (3 REQ.)
3. C364LT - STRINGER (12 REQ.)
4. ZEE SECTION STIFFENER - 24 ST
5. 52-31200-8 - FORMER
6. 55-31038-4 - DOOR ASSEMBLY
7. 66-31009 - VENTILATOR ASSEMBLY
8. 52-31200-5 - FORMER - AT-6A - AT-6B
   66-31200 - FORMER - AT-6C
9. 52-31200-5 - FORMER - AT-6A - AT-6B
   66-31201 - FORMER - AT-6C
10. 58-31200-1 - FORMER - AT-6A AT-6B
    52-31200-3 - FORMER - AT-6C

Figure 31—Front Fuselage Side Panels

36. CONSTRUCTION OF FRONT FUSELAGE SIDE PANELS.

The side panels cover the bay between the upper and lower longerons of the front tubular section of the fuselage. The panels consist of vertical formers and longitudinal stringers covered by means of .025 and .032 inch thick 24ST alclad sheeting *(see Figure 31)*. The stringers are all C364LT rolled type and the upper and lower Z sections are formed from .025 inch thick 24ST alclad sheet. The assembly is spot-welded about the outside edges and the remaining structure is assembled by rivets. The panels are attached by means of screws to supports located about the outside edges of the bay. The side panels are not designed to carry any load and serve only as fairings.

37. FORWARD FORMER OF FRONT FUSELAGE SIDE PANELS.

The single forward former on both the left and right fairing side panels of the front fuselage may be spliced as shown *(see Figure 32)*. If the damage is extensive, cut out the damaged

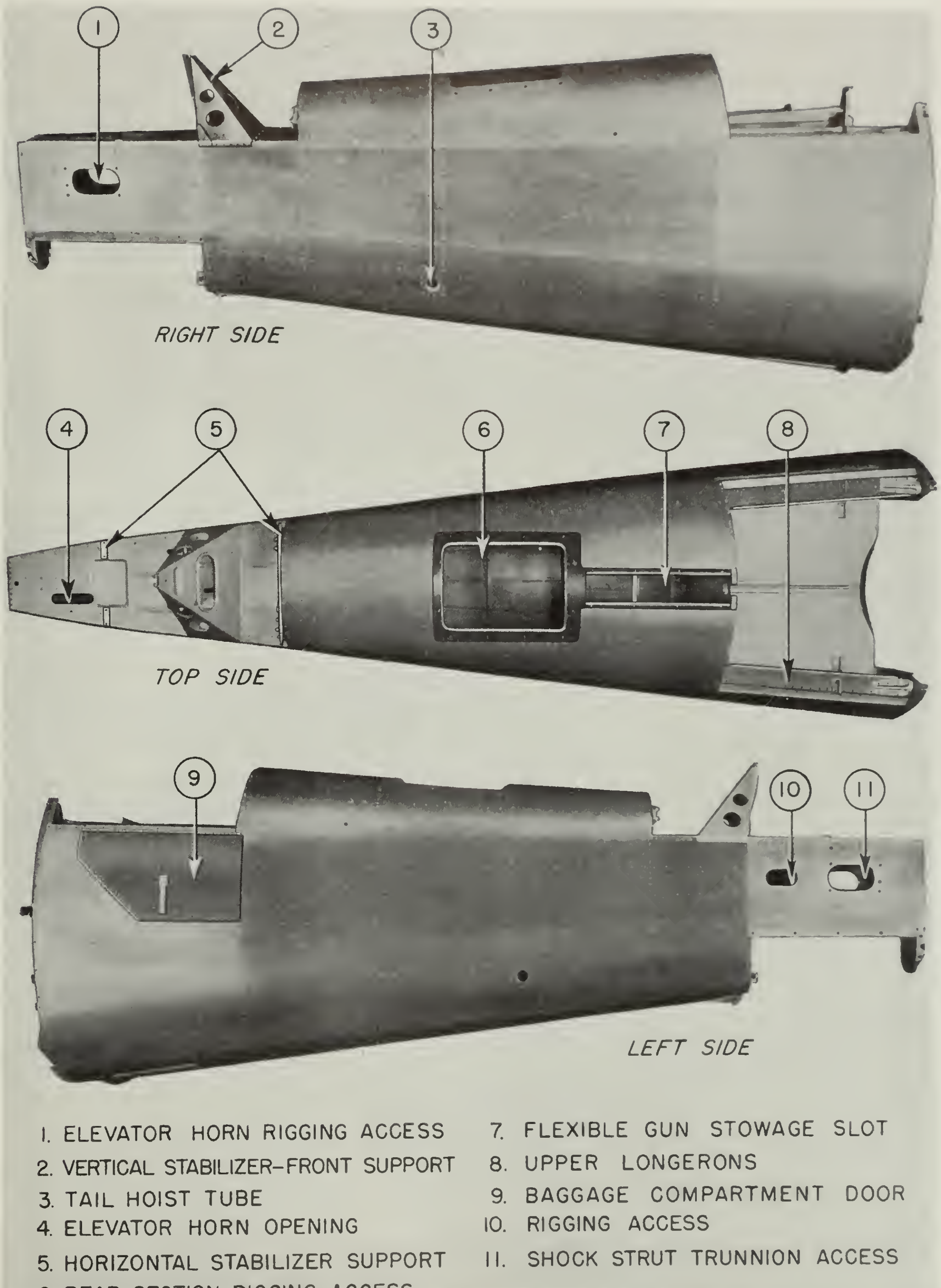
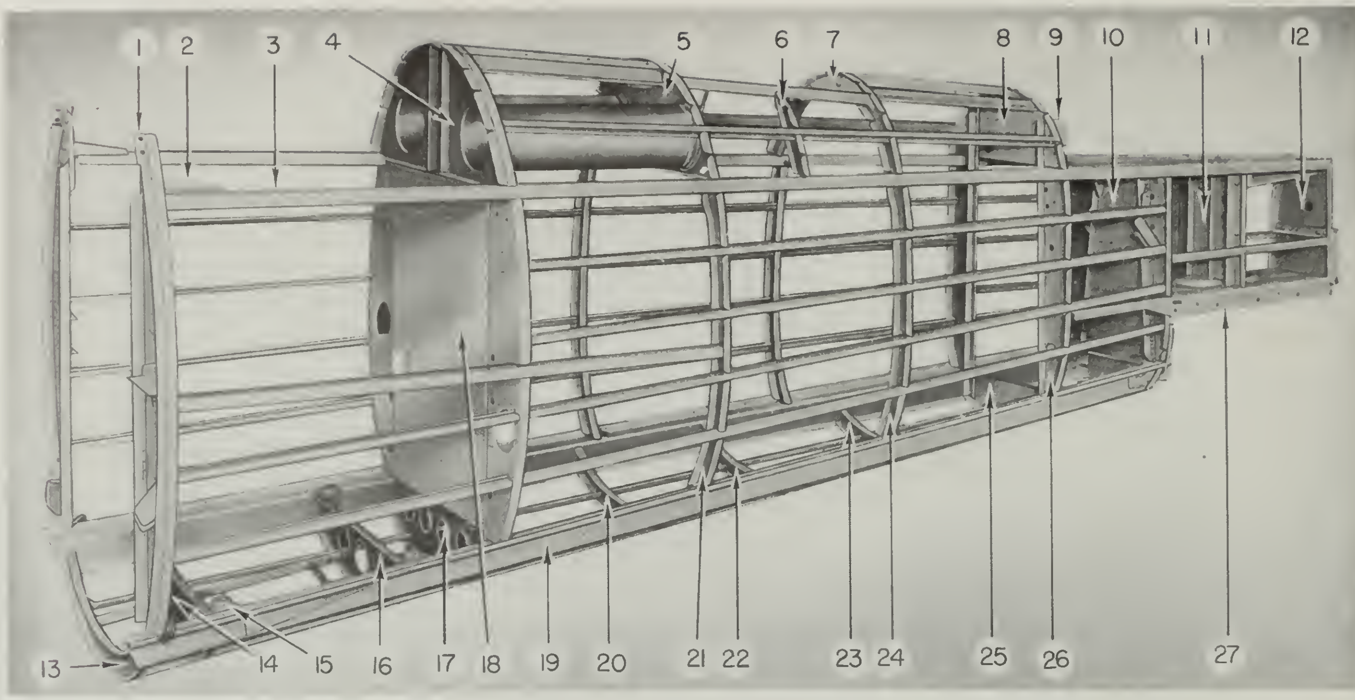


Figure 33—Aluminum Rear Fuselage Covered Assembly



NORTH AMERICAN AVIATION PART NUMBERS

	CONTRACT NOS. AT-6A	AT-6B	AT-6C	DRAWING NOS.	TITLE
1.	77	77	88	31113	BULKHEAD - STATION 106.5 R.H.
2.	52			31142	FITTING - FUSELAGE ATTACHING L.H.
3.	58			31122	LONGERON - UPPER L.H.
4.	55*				
	77*	78		31117	BULKHEAD - STATION 137 UPPER
5.	55			31391-3	BULKHEAD - STATION 158-1/4 UPPER
6.	55			31416	FORMER - STATION 168-1/2 UPPER LEFT
7.	77*			31119	BULKHEAD - STATION 179-1/2 UPPER
8.	69			31120	BULKHEAD - STATION 200-3/4 UPPER
9.	52			31109	FITTING - HORIZONTAL STABILIZER ATTACHING
10.	66			31114-12	BULKHEAD - STATION 215-5/8 REAR
11.	56			31115-6	BULKHEAD - STATION 227-3/8 REAR
12.	55			31116	BULKHEAD - STATION 242-3/16 UPPER
13.	72			31146	FORMER - STATION 106-1/2 LOWER
14.	55			31169-6	FORMER - FUSELAGE REAR SECTION 1B LOWER
15.	36			31289	MOUNT, CAMERA L.H.
16.	77		77*	31170	FORMER - FUSELAGE REAR SECTION 1B LOWER
			88*	31497	FORMER - FUSELAGE REAR SECTION 1B LOWER
17.	64		64*	31245	BULKHEAD - STATION 137 LOWER
			88*	31496	BULKHEAD - STATION 137 LOWER
18.	77			31112	BULKHEAD - STATION 137
19.	77			31121	LONGERON - LOWER L.H.
20.	77		77*	31395	FORMER - STATION 156-11/16 LOWER
			88*	31495	FORMER - STATION 156-11/16 LOWER
21.	50			31121	FORMER - STATION 158-1/4 LEFT SIDE
22.	55*			31396	FORMER - STATION 170-3/8 LOWER
23.	55			31397	FORMER - STATION 185-9/16 LOWER
24.	50			31122	FORMER - STATION 179-1/2 LEFT SIDE
25.	55			31398	FORMER - STATION 200-3/4 LOWER
26.	56			31123	FORMER - STATION 200-3/4 LEFT SIDE
27.	55			31402	CHANNEL - ASSEMBLY

* *INDICATES CODED PART*

Figure 34—Aluminum Rear Fuselage Part Numbers

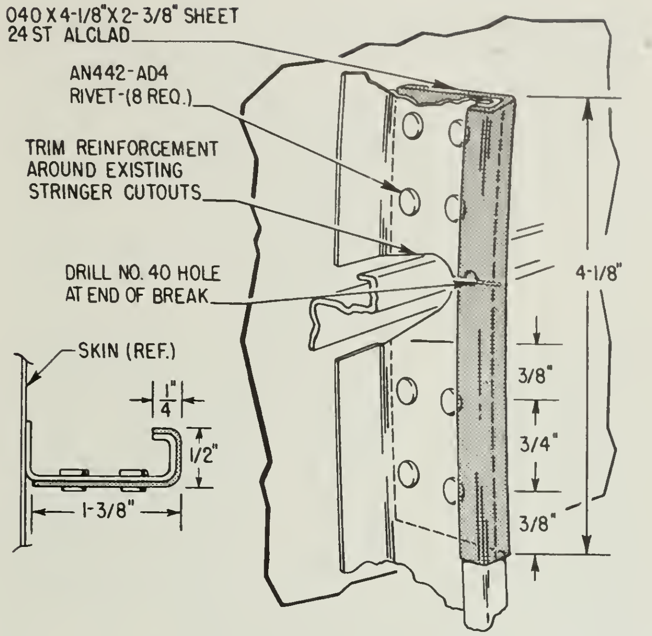


Figure 35—Splice of Rear Four Formers of Front Fuselage Side Panels

material with a hack saw. To serve as the splice members, cut two sheets of .040 inch thick 24ST alclad having a length equal to the damage plus 2-3/4 inches and a width of approximately 1-1/2 inches. Along the length of one of the splice sheets bend a 90-degree, 1/2-inch flange. Along the length of the other splice sheet, bend a 90-degree, 5/8-inch flange. Trim the splice sheets to the proper shapes. If there are any skin rivets in the affected area, drill them out. Clamp the splice members to the damaged former. Center punch the required eight rivet locations in the noted pattern, and drill the center-punched rivet locations with a No. 30 (.1285) drill. Cut countersink the rivet holes 100 degrees by 7/32-inch. Remove the splice members, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the members to the damaged former. Into the rivet holes, insert and drive eight AN426-AD4 rivets.

38. REAR FOUR FORMERS OF FRONT FUSELAGE SIDE PANELS.

Any of the rear four formers of the front fuselage side panels may be repaired as shown *(see Figure 35)*. If the damage is in the form of a crack, drill a No. 40 (.098) hole at the end of the crack. For the splice member, cut a sheet of .040 inch thick 24ST alclad having an approximate length of 4-1/8 inches and a width of 1-3/8 inches. Along the length of the splice member, bend up a 1/4-inch, 90-degree flange; and then bend up a 1/2-inch, 90-degree flange measured from the first bend. Observe a minimum bend radius of 1/8-inch. Trim the splice member to match any existing stringer cutouts in the affected area. With a No. 40 (.098) drill, drill out the affected skin rivets at each side of the damage. Clamp the splice member to the proper position on the damaged former. Center punch the rivet locations in the noted pattern. Drill the center-punched rivet locations with a No. 30 (.1285) drill. Remove the splice member, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the splice member to the damaged former; and into the eight rivet holes, insert and drive AN442-AD4 rivets.

39. ALUMINUM REAR FUSELAGE CONSTRUCTION.

The semimonocoque rear fuselage section is 11 feet 3.22 inches in length and is constructed wholly of 24ST alclad, consisting essentially of four longeron formers and thin skin stiffened with stringers *(see Figures 33 and 34)*. The longerons extend the entire length of the structure and form the upper and lower points of attachment to the forward fuselage truss assembly *(see Figure 36)*. The upper and lower

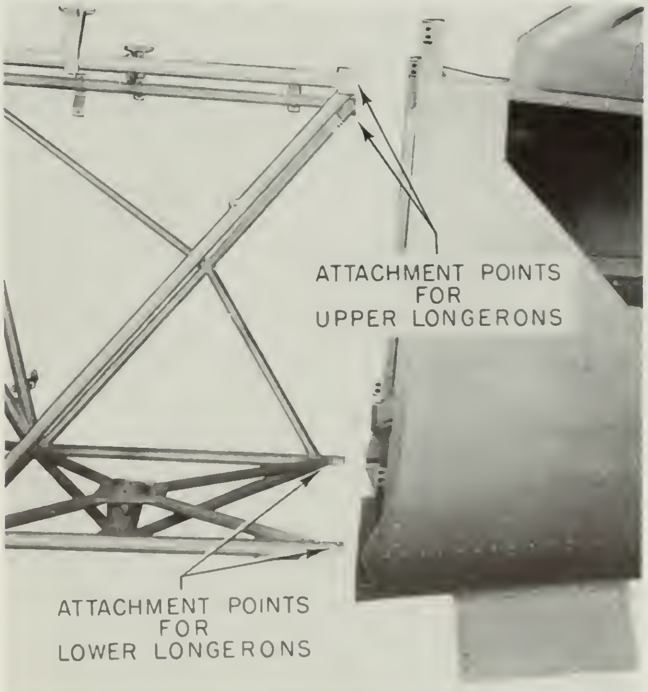
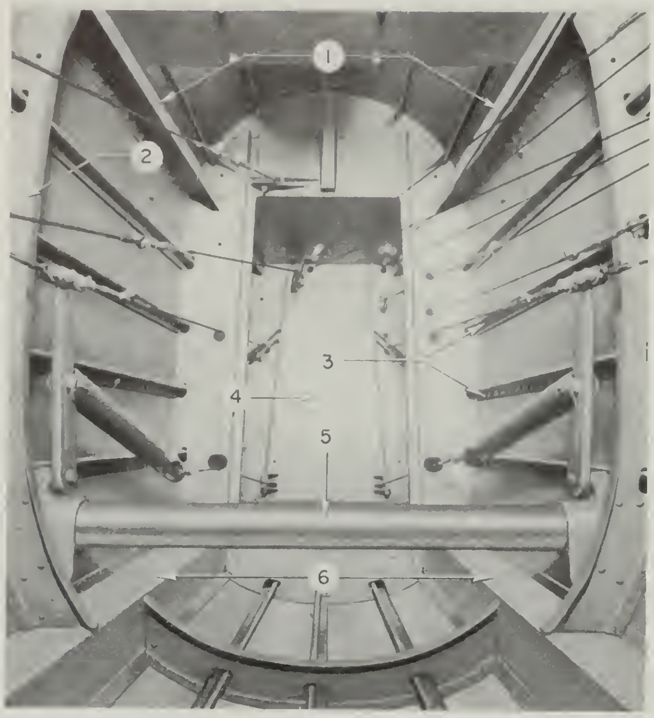


Figure 36—Attachment of Front Truss to Aluminum Rear Fuselage



1. UPPER LONGERONS
2. FUSELAGE FRAME
3. SKIN STRINGERS
4. BULKHEAD
5. LIFT TUBE
6. LOWER LONGERONS

Figure 37—Aluminum Rear Fuselage Interior

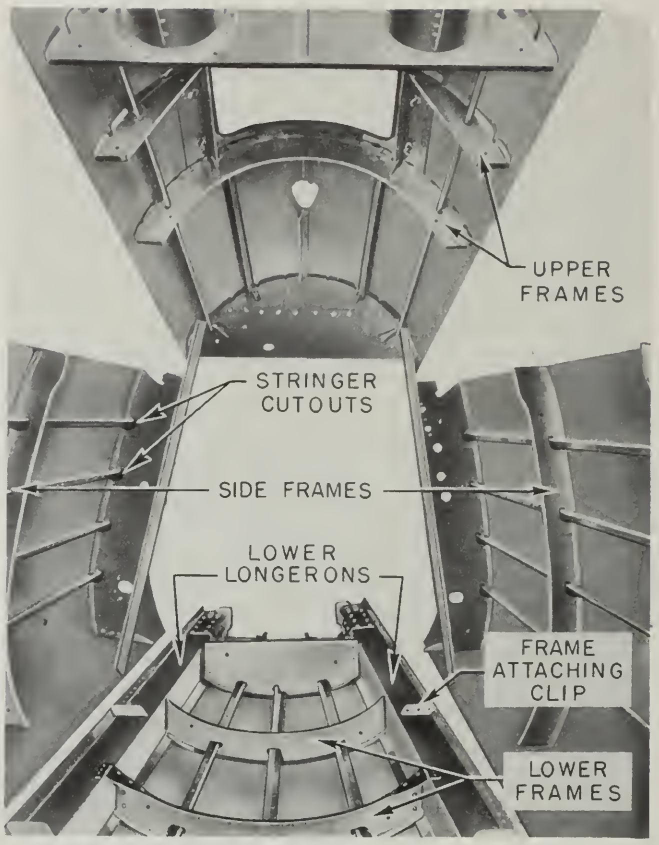


Figure 38—Aluminum Rear Fuselage Interior Exploded

longerons are designed to resist the main bending loads. A torsionally rigid structure is formed by the skin structure and the main bulkheads *(see Figure 34)*. The light, rolled-sheet stringers serve only to stiffen the skin and to prevent undue distortion and buckling. Inasmuch as the rear fuselage section is semi-monocoque, the outer skin resists all shear.

40. ALUMINUM REAR FUSELAGE FORMERS.

A typical repair which may be used for all the rear fuselage formers is illustrated *(see Figure 39)*. If the damage is in the form of a crack, perhaps at a stringer cutout, drill a No. 40 (.098) hole at the end of the crack. For the splice members, cut two sheets of .040 inch thick 24ST alclad, each having a length of 5-3/4 inches and a width of 2-3/4 inches. Along the length of one of the splice members, bend an 11/16-inch flange. Along the length of the other splice member, bend a 13/16-inch flange. Observe a minimum bend radius of 1/8-inch. Trim the splice members around any existing cutouts in the damaged former. Drill out the affected skin rivets at each side of the damage, using a No. 40 (.098) drill. Clamp the splice members to the damaged former. Center punch the required sixteen rivet locations in the noted pattern. With a No. 30 (.1285) drill, drill the center-punched rivet locations and drill the splice members through the existing rivet holes in the skin. Observe a minimum edge distance of 3/8-inch measured from the center of the rivet hole to the edge of the material involved. Remove the splice members, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the splice members to the damaged former. Through the holes in the skin, insert and drive B1227-AD4 rivets in the quantity required. Into the remaining rivet holes, insert and drive sixteen AN442-AD4 rivets *(see Figure 39)*.

41. REPLACING ALUMINUM FORMERS.

If replacement of a damaged former is warranted, a duplicate spare part should be utilized for the replacement. The replacement part numbers of the front fuselage fairing panels are shown *(see Figure 31)*. The part numbers of the aluminum rear fuselage formers are also shown *(see Figure 34)*. Where immediate replacement is necessary and a spare part is not available, replacement formers may be fabricated locally, using the damaged former as a template. The procedure may be summed up as follows: Lay out the shape of the affected former upon a panel of 1/2-inch wood. In several places, temporarily nail this panel to a second panel of 1/2-inch wood. On a band saw or with a keyhole saw, cut around the marked contour. Pull out the nails and separate the two equivalent panels cut to the contour of the affected former. Cut a sheet of 24ST alclad of the same thickness and developed width and length as the damaged former. If any stringer cutouts are present in the damaged former, make cuts in the replacement sheet to match. Clamp the sheet of alclad between two wood form blocks in a vise; and with a rawhide mallet, pound the flanges of the replacement former over the edge of the wood form blocks *(see Figure 40)*. Remove the former from the vise and make any further modifications that may be necessary, such as fairlead holes, etc. Smear all overlapping surfaces of the replacement former with zinc chromate primer. Install the replacement former in exactly the same manner as the original.

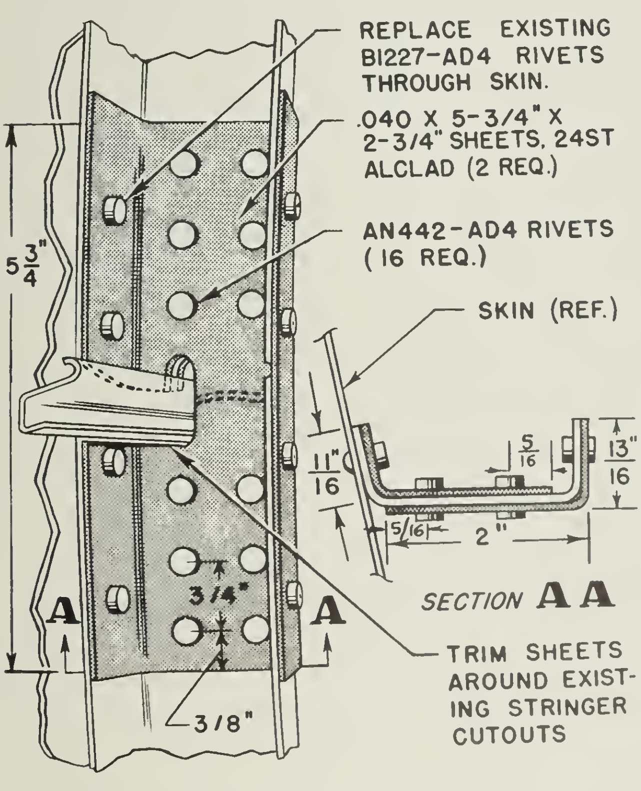


Figure 39—Aluminum Rear Fuselage Former Splice

42. ALUMINUM STRINGER TYPE C107LT-40.

The Type C107LT-40 stringer is used in several locations of the rear fuselage lower covered frame assembly on the AT-6C Airplanes only. The stringer is formed of rolled 24ST alclad sheet .040 inch thick having no equivalent Alcoa die number. If the damage is slight in nature, a C107LT-40 splice member 6-1/2 inches long will suffice *(see Figure 41)*. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut midway between two skin rivets at each side of the damage. For the splice member, cut a length of C107LT-20 stringer equal to 6-1/2 inches plus the length of the damage. Skin rivets at the end of the splice must have at least 3/8-inch edge distance and this may necessitate making the splice member slightly longer than 6-1/2 inches plus the damage length. If the prepared C107LT-20 stringer material is not available, bend up an angle locally from .040 inch thick 24ST alclad sheet material to serve as the splice member. Drill out the affected skin rivets at each side of the damage, and clamp the stringer splice member to the damaged stringer. With a No. 40 (.098) drill, drill the splice member through the existing skin rivet holes. If the existing skin rivets are spaced at 1-1/4 inches on centers, or greater, double the number of existing skin rivet holes at the splice. With a No. 30 (.1285) drill, drill the five holes at 5/8-inch on centers through the stringer upstanding flange and the

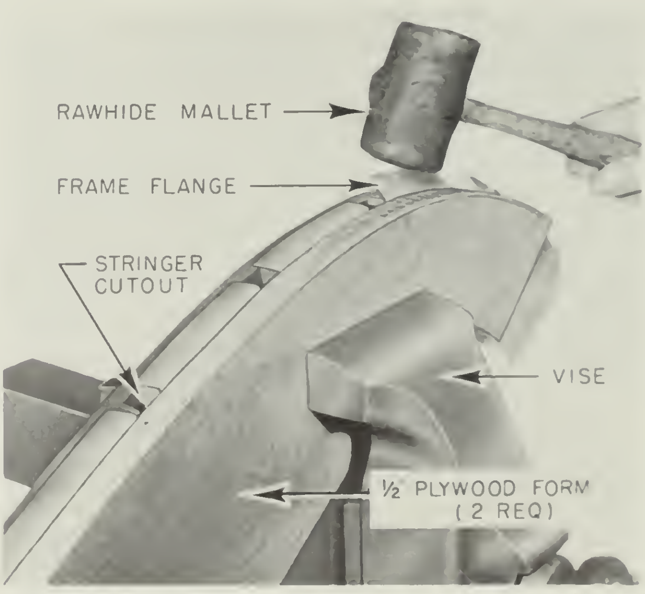


Figure 40—Fabricating Replacement Formers

splice member at each side of the damage. Remove the splice member, and burr the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice member. Clamp the splice member to the damaged stringer. At each side of the damage, drive the five AN442-AD4 rivets in the upstanding flange, and rivet the splice member through the skin with the same size and type rivets previously removed from the skin *(see Figure 41)*.

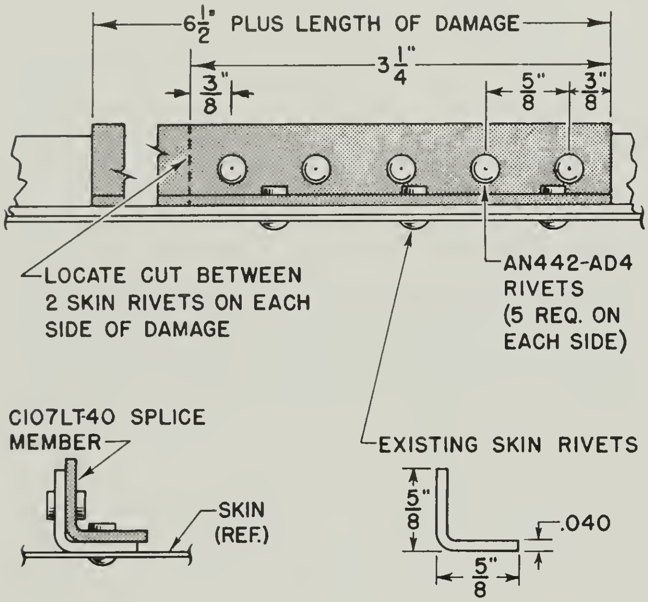


Figure 41—Splice for Stringer Type C107LT-40

43. ALUMINUM STRINGER TYPE C180T.

The Type C180T stringer is used in the construction of the fuselage rear section upper deck assembly. The stringer is a bulb angle of extruded 24ST aluminum alloy material, having the Alcoa type designation of K14280. The longest length of the stringer employed in any location is 21 inches; therefore, if damage occurs to this type stringer, replace the entire damaged stringer section.

44. ALUMINUM STRINGER TYPE C234LT.

The stringer Type C234LT is used extensively in the construction of the rear aluminum fuselage. The stringer is a Z section of .025 inch thick 24ST alclad, having no equivalent Alcoa die number. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut midway between two skin rivets at each side of the damage *(see Figure 42)*. Skin rivets at the end of the splice must have at least 3/8-inch edge distance, and this may necessitate making the splice member slightly longer than that noted. Clamp the stringer splice member to the damaged stringer. With a No. 21 (.159) drill, drill five rivet holes at each side of the damage at 5/8-inch on centers after center punching the hole locations. Drill the splice member through the skin with the same size drill and spacing as the existing skin rivets through the damaged stringer. Remove the splice member, and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice member. Again clamp the splice member to the stringer. At each side of the damage through the stringer upstanding flange, insert and drive five AN442-AD5 rivets. Through the skin and the splice member, insert and drive B1227-AD rivets of the size and quantity required *(see Figure 42)*.

45. ALUMINUM STRINGER TYPE C364LT.

The stringer Type C364LT is used in the construction of the front fuselage fairing side panels. The stringer is formed of .020 inch thick 24ST alclad, having no equivalent Alcoa die number. If the damage is greater than a crack, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice member, cut a sheet of .032 inch thick 24ST alclad 2 inches wide and having a length of 7-1/2 inches plus the length of the damage *(see Figure 44)*. Along the length of the splice sheet, bend a 1/4-inch, 90-degree flange; and then make a 90-degree, 7/16-inch bend measured from the first flange. Along the opposite edge of the length of the splice sheet, bend a 90-degree, 1/2-inch flange. Use a hand brake if available and observe a minimum bend radius of 1/8-inch. With a No. 40 (.098) drill, drill out the affected skin rivets at each side of the damage. Clamp the splice sheet to the damaged stringer, and drill the splice member through the existing skin rivet holes, using a No. 40 (.098) drill. Double the number of skin rivet locations at the splice location if the existing rivet holes in the skin are spaced greater than 1-1/4 inches on centers. In the upstanding flange of the splice member, center punch five rivet locations at each side of the damage at an average spacing of 3/4-inch on centers. Drill the center-punched rivet locations with a No. 30 (.1285) drill. Remove the splice members, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice sheet. Again clamp the splice sheet to the damaged stringer.

Replace the skin rivets with rivets of the same size and type as previously removed. In the remaining rivet holes, insert and drive ten AN442-AD4 rivets *(see Figure 44)*.

46. ALUMINUM STRINGER TYPE C366T.

The stringer Type C366T is used in several locations in the lower section of the rear fuselage. The stringer is a 24ST aluminum right-angled bulb section having the equivalent Alcoa die number of L24910. The longest length of the extruded section employed in the construction of the rear fuselage is 23 inches; therefore, if damage occurs to this type stringer, replace the entire damaged section.

47. ALUMINUM REAR FUSELAGE BAGGAGE COMPARTMENT INTERCOSTAL ANGLE.

The NA 55-31401 intercostal angle located on the left side of the rear fuselage below the baggage compartment may either be replaced or spliced if damaged. The intercostal angle extends 51 inches aft from the fuselage monocoque joining surface and is formed of J-shaped, .081 inch thick, subsequently heat-treated 24SO alclad. The splice of the member is shown *(see Figure 43)*. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut between two skin rivets at each side of the damage. For the splice member, cut a sheet of .081 inch thick 24SO alclad having a width of 2-5/16 inches and having a length of 9 inches plus the length of the damage. Along the length of the splice member, bend up a 90-degree, 5/16-inch flange; and then bend up a 90-degree, 1-3/32 inch flange measured from the first bend. Use a hand brake to bend the sheet, and observe a minimum bend radius of 5/32-inch. After the 24SO sheet is formed to the proper shape, heat treat the member to the full hardness of 24ST alclad as outlined in Section I. If any rework on the splice sheet is necessary, accomplish the work within the first hour after the material is heat treated, as the material may be worked with comparative ease during this period. Within a distance of 4-1/2 inches on each side of the damage, drill out the existing rivets in the damaged intercostal angle. Clamp the splice member to the damaged intercostal angle; and with a No. 30 (.1285) drill, drill the splice member through the existing rivet holes in the skin. At each side of the damage along the top flange of the intercostal angle, center punch seven rivet locations at 5/8-inch on centers, and observe a minimum edge distance of 3/8-inch measured from the center-punch mark to edge of the material involved. If the damage is located in the baggage compartment where the top flange of the intercostal angle is riveted to the apron assembly of the baggage compartment, double the number of the rivet locations through the apron assembly and the intercostal angle at the splice location. With a No. 21 (.159) drill, drill the fourteen center-punched rivet locations along the top flange of the inter-

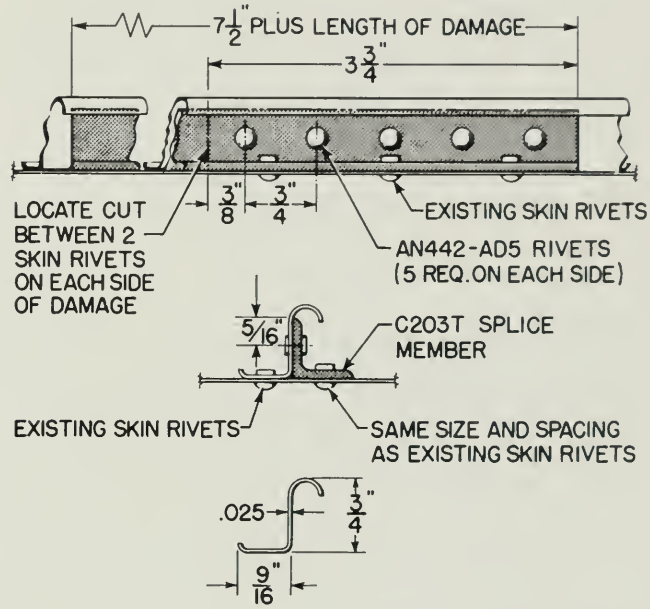


Figure 42—Splice for Stringer Type C234LT

costal angle. Remove the splice member, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the splice member to the damaged intercostal angle; and through the skin rivet holes, insert and drive B1227-AD4 rivets in the quantity required. Through the rivet holes in the top flange of the intercostal angle, insert and drive fourteen AN442-AD5 rivets *(see Figure 43)*. If the damage is located in the baggage compartment, rivet the top flange of the intercostal angle through the baggage compartment apron assembly with B1227-AD5 rivets.

48. ALUMINUM REAR FUSELAGE UPPER LONGERONS.

The rear fuselage upper longerons which extend 120 inches aft of the fuselage joining surface on the left and the right side of the fuselage is formed of a bent-up U-shaped sheet of .064 inch thick 24ST alclad. A damaged longeron may be replaced, but repair is quite simple and is preferable to replacement. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut between two skin rivets at each side of the damage. For the splice members, cut two sheets of .064 inch thick 24ST alclad, each 3 inches wide and having a length of 9 inches plus the length of the damage *(see Figure 46)*. Along the length of one of the splice members, bend a 90-degree, 1-1/16 inch flange. Along the length of the other splice member, bend a one-inch flange to match the open angle of the damaged longeron. Use a hand brake to bend the splice members, and observe a minimum bend radius of 3/16-inch. With a No. 40 (.098) drill, drill out the affected skin rivets in the damaged longeron for a distance of 4-1/2 inches on each side of the damage. Locate the drill in the depression of the skin rivet, and apply the drill power in short bursts until the rivet head twists from the shank; then punch out the rivet. If the damage is located on the right side of the fuselage between the first and second formers aft of the fuselage joining surface, drill out the affected rivets securing the gunner's hood cowling attaching nut plates to the skin side flange of the longeron. If the damage is located on the left side of the fuselage between the first and second formers aft of the fuselage joining surface, drill out the cowling nut plates; and in addition, drill out the rivets securing the baggage compartment hinge to the skin side flange of the longeron. Clamp the splice members to the proper positions on the damaged longeron *(see Figure 46)*. With a No. 30 (.1285) drill, drill the splice member flange through the existing skin rivet holes. If the damage is located between the first and second formers aft of the fuselage joining surface, drill the one splice member through the existing skin rivet holes with the same size drill as the existing holes. Center punch the 36 splice rivet locations at an average spacing of 3/4-inch on centers, observing a minimum edge distance of 3/8-inch measured from the center-punch mark to the edge of the material involved *(see Figure 46)*. With a No. 21 (.159) drill, drill the 36 center-punched rivet locations. Remove the splice members, and burr all rivet holes. Apply one coat of zinc chromate to all overlapping surfaces of the splice members. Again clamp the splice members to the proper positions on the damaged longeron. Through the skin rivet holes, insert and drive B1227-AD4 rivets in the quantity required. If the

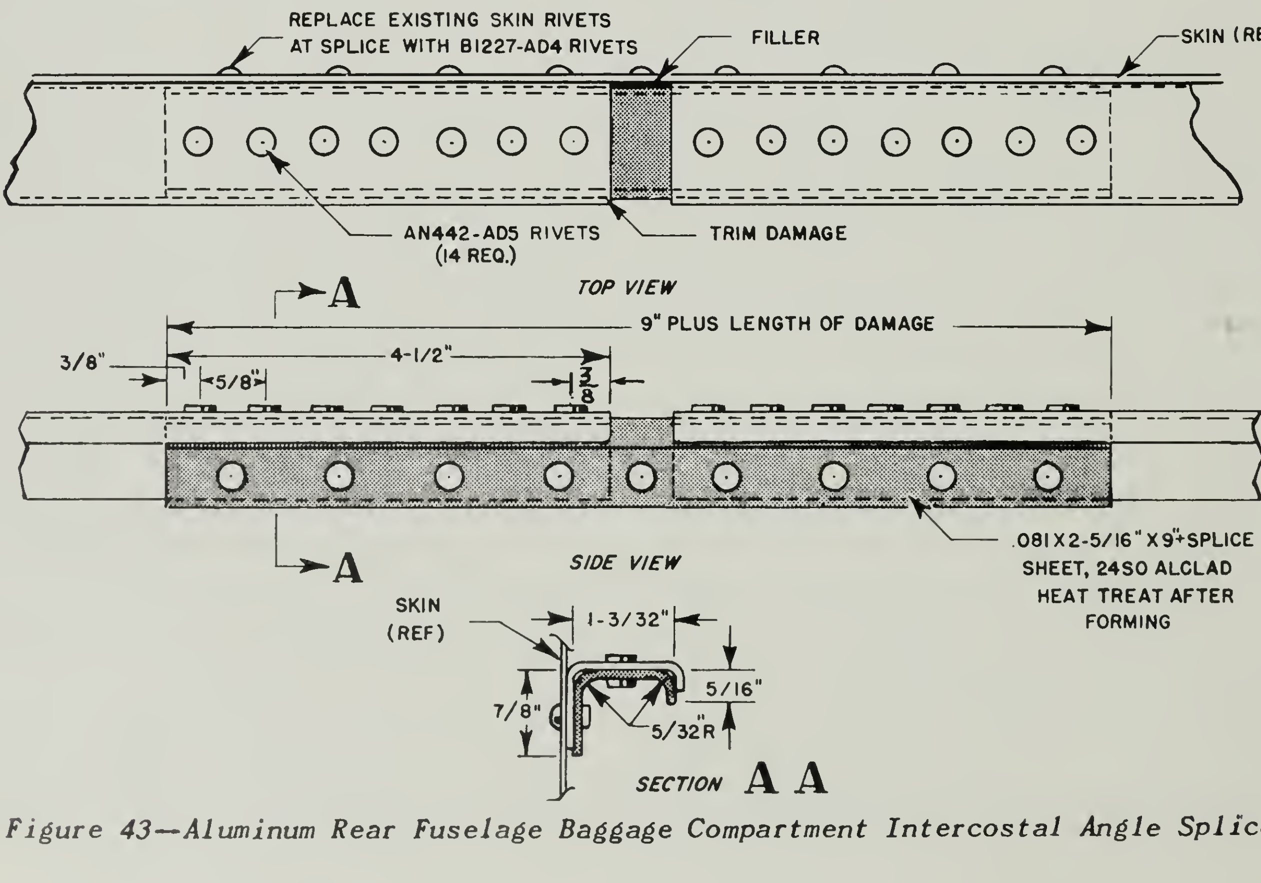


Figure 43—Aluminum Rear Fuselage Baggage Compartment Intercostal Angle Splice

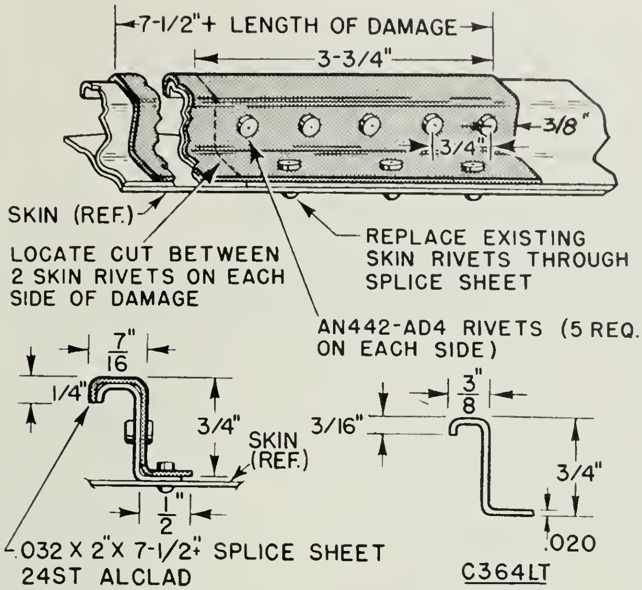


Figure 44—Splice for Stringer Type C364LT

Figure 45—Aluminum Rear Fuselage Doors

damage is located on the right side between the first and second frames aft of the fuselage joining surface, rerivet the existing nut plates to the skin side flange of the longeron with AN426-AD3 rivets. If the damage is located on the left side of the fuselage between the first and second frames aft of the joining surface, in addition to reriveting the nut plates, rerivet the baggage compartment hinge to the longeron with AN426-AD4 rivets in the quantity required. Into the remaining 36 rivet holes through the splice members and the damaged longeron, insert and drive AN442-AD5 rivets *(see Figure 46)*.

49. ALUMINUM REAR FUSELAGE LOWER LONGERONS.

The lower longerons extending 111 inches aft of the fuselage joining surface on the left and right side of the rear fuselage may be spliced as shown *(see Figure 47)*. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut between the skin rivets at each side of the damage. On the upper flange of the damaged longeron, drill out the skin rivets for a distance of 3-3/4 inches on each side of the damage and remove any existing former attaching angles. On the lower flange of the damaged longeron, drill out the skin rivets along the entire length of affected skin panel and bend the skin panel upwards to gain access to the inside of the longeron at the affected area. Use a No. 40 (.098) drill and place the drill into depression in the rivet head; apply the drill power in short bursts until the rivet head twists from the shank; then punch out the remaining portion of the rivet. Use this method to remove all skin rivets, and take particular care to prevent the elongation of any of the existing rivet holes. For the splice member, cut a sheet of .051 inch thick 24ST alclad having a width of 7 inches and having a length of 7-1/2 inches plus the length of the damage. Along the length of the splice sheet, bend a 5/8-inch flange to an angle matching the upper longeron angle. Apply the splice member to the top surface of the damaged longeron so that the bent-up splice member flange fits the inside radius of the longeron flange. At the intersection of the top and side surfaces of the longeron, make a grease pencil mark along the length of the splice member. The distance between the bend of the first 5/8-inch flange and the marked line should be between 2-3/16 and 2-3/4 inches. Along this line brake a 90-degree flange. Measure the distance on the damaged longeron between the intersection of the top and side of the longeron to the beginning of the bend in the lower flange of the longeron. Mark this distance on the splice member and on a hand brake; bend the splice member flange along this line to match the bottom flange on the longeron. Trim the excess material from the bottom flange of the splice member, so that the bottom flange has a constant

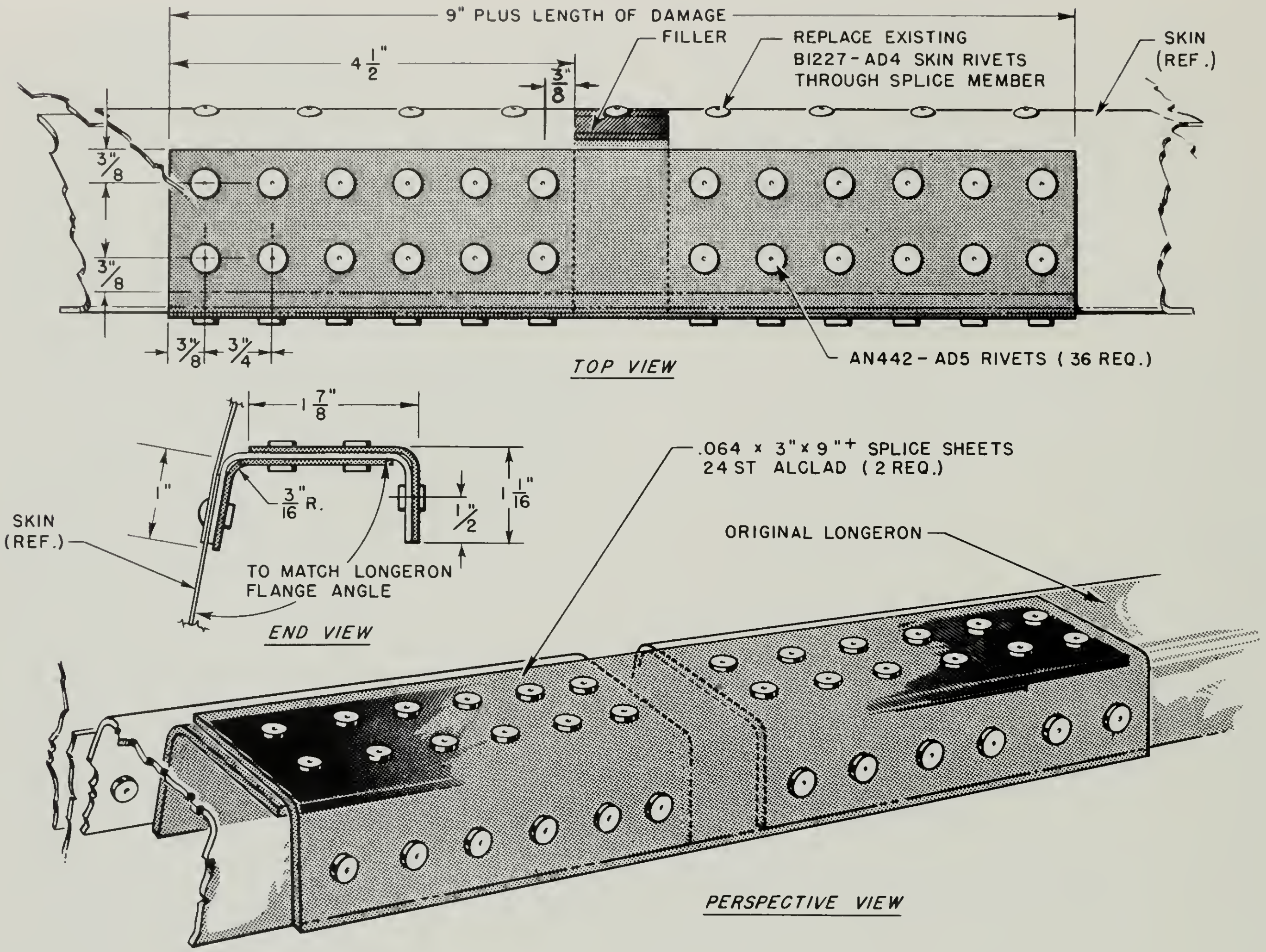


Figure 46—Aluminum Rear Fuselage Upper Longeron Splice

width of 5/8-inch. In all of the bending operations on the splice member, observe a minimum bend radius of 3/16-inch. Apply the splice member to the damaged longeron so that the splice member overlaps the damage on each side by 3-3/4 inches. At each side of the damage, on the top and side surface of the splice member, lay out with a pencil two rows of rivet locations in the noted pattern *(see Figure 47)*. Center punch the 40 rivet locations; and with a No. 21 (.159) drill, drill the splice member through at opposite ends and fasten the member in place with skin fasteners. Drill the remaining center-punched rivet locations with the No. 21 (.159) drill. Remove the splice member, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice member. Again secure the splice member to the longeron with skin fasteners. Through the 40 rivet holes in the splice member, insert and drive AN442-AD5 rivets. To provide access for bucking the rivets on the inside of the longeron, bend the skin panel upward at the lower flange of the longeron, the securing skin rivets having been previously drilled out. Rerivet to the longeron any previously removed former attaching angles. On the entire length of the skin edge at the lower flange of the longeron, replace the previously drilled-out skin rivets with B1227-AD4 rivets except in the immediate longeron repair area. With a No. 30 (.1285) drill, drill the upper and lower splice member flanges through the existing skin rivet hole locations. Using a thin hook-shaped hack saw blade with the teeth removed, scrape out the drill chips between the sheets at the skin rivet locations. At the splice location, replace

BI227-AD4 skin rivets through the splice member, bucking the rivets from the interior of the fuselage *(see Figure 47)*.

50. ALUMINUM SKIN AND BULKHEAD WEBS.

The skin arrangement of the fuselage consists principally of 24ST alclad aluminum alloy *(see Figure 48)*. The bulkhead webs of the rear fuselage are also formed of thin 24ST alclad. For the repair of the skin and the bulkhead webs, see the pertinent paragraphs applicable to skin in Section III. Inasmuch as access to the most of the fuselage interior may be readily gained, all fuselage skin patches should be applied to the interior. On the inner or outer skin of the baggage compartment door and camera door *(see Figure 45)*, any patching may be accomplished with LSI127-4-2 Cherry blind rivets. The double skin construction of these doors provides no interior access for rivet bucking.

51. FAIRINGS AND FILLETS.

On the earlier AT-6C Series Airplanes and previous North American Aviation, Inc., trainer aircraft, the fairings and fillets are made from 3S aluminum alloy, 52S aluminum alloy, and subsequently heat-treated 24SO alclad *(see Figure 48)*. For the repair of the alclad fairings and fillets, see the pertinent paragraphs on skin repairs in Section III. Patching 3S and 52S aluminum alloy may be accomplished by welding equivalent material over the hole, using a torch and a 2S aluminum alloy welding wire. On later AT-6C Series Airplanes, some of the aluminum fairings and fillets are replaced with plastic fillets of phenolic sheet and Chemold. These plastic fairings and fillets are considered expendables; and as such, they should be replaced rather than repaired *(see Figure 49)*.

52. EXTRUDED ALUMINUM REPAIR MATERIAL.

The extruded cross-sectional shapes that are required in the repair of the structural members of the fuselage are listed below. The material used in forming these extrusions is 24ST aluminum alloy conforming to Federal Specification QQ-A-354. Numbers with a C prefix are North American Aviation, Inc., standard parts. Following the North American Aviation, Inc., part number, the Aluminum Company

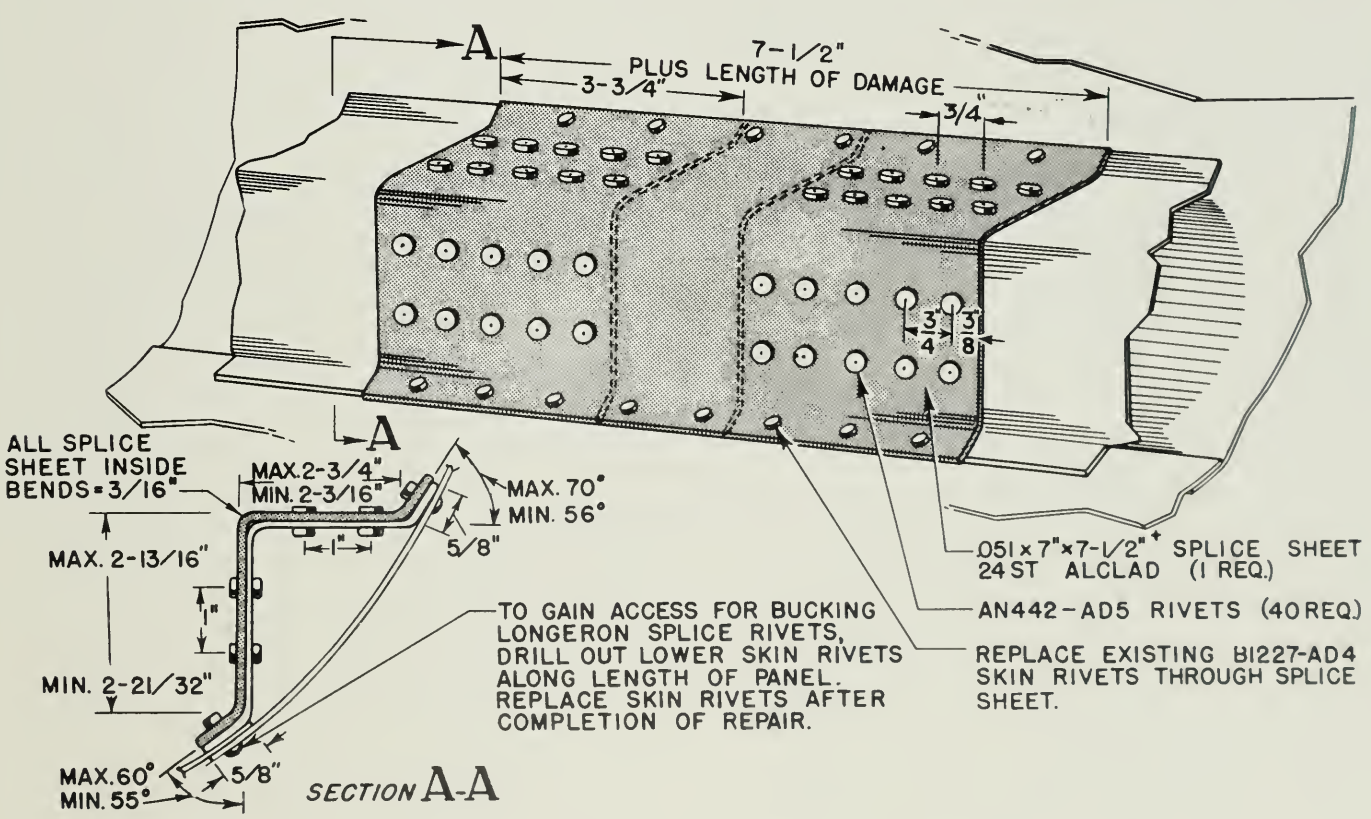
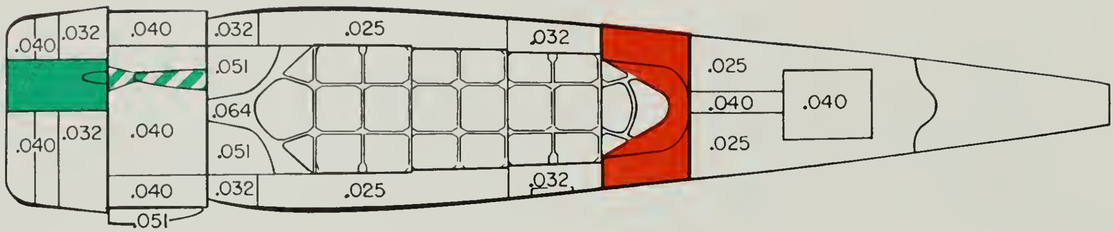
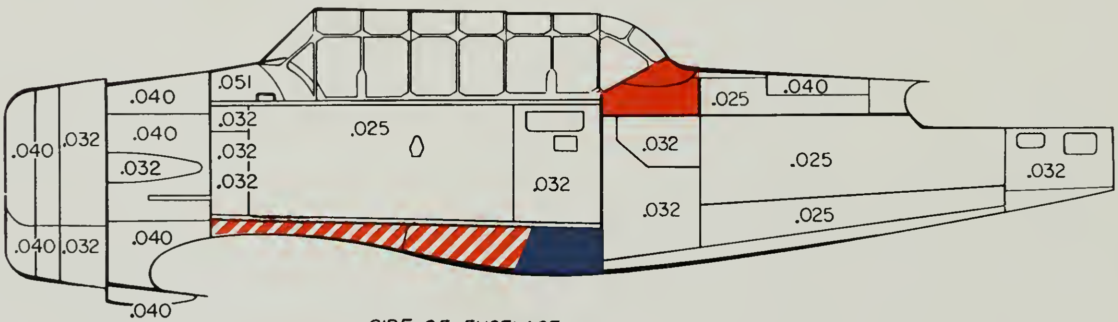


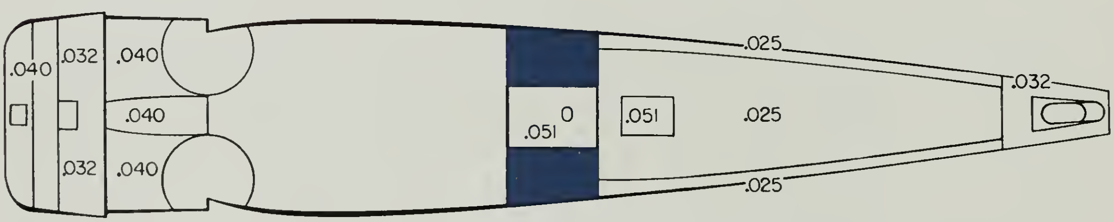
Figure 47—Aluminum Rear Fuselage Lower Longeron Splice



TOP OF FUSELAGE



SIDE OF FUSELAGE



BOTTOM OF FUSELAGE

CODE

- . 040 3S AL. ALLOY
- . 051 52S AL. ALLOY
- . 038 CORR. RESIST. STEEL
- . 032 52S AL. ALLOY
- . 025 CORR. RESIST. STEEL

NOTE: ALL SKIN IS 24 ST ALCLAD EXCEPT AS NOTED AND DECIMAL FRACTIONS INDICATE SKIN THICKNESS IN INCHES

Figure 48—Fuselage Skin Arrangement

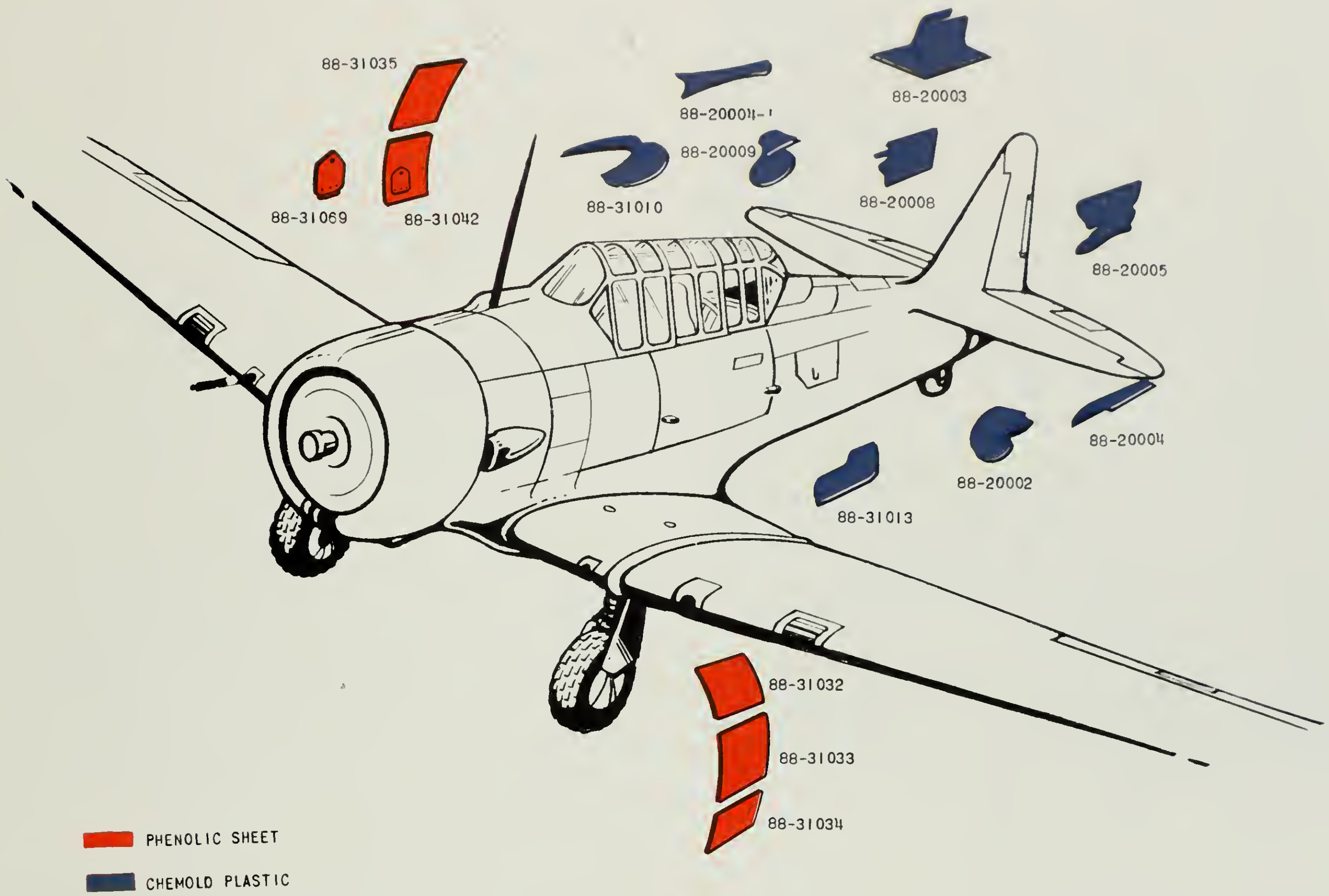


Figure 49—Plastic Fairings and Fillets

of America (Alcoa) equivalent die number is listed in parentheses.

EXTRUSION DIE NUMBERS

C180T	(K14280)
C366T	(L24910)

53. ALCLAD SHEET REPAIR MATERIAL.

The aluminum sheet material used in the repair of the fuselage should be 24ST alclad conforming to Federal Specification QQ-A-362. The various thicknesses of sheets that may be required are listed below. The lengths and widths of the sheets must be determined locally for the extent of repairs to be undertaken, with reference to the repair procedures outlined in the appropriate paragraphs of this Section.

THICKNESS OF 24ST ALCLAD (INCHES)	REMARKS
.020	SKIN, BULKHEADS (C364LT)
.025	SKIN, BULKHEADS (C234LT)
.032	SKIN, BULKHEADS, FORMERS
.040	SKIN, FORMERS, (C107LT-40)
.051	SKIN, LONGERONS
.064	LONGERONS
.081	INTERCOSTAL

54. RIVETS REQUIRED FOR ALUMINUM REPAIRS.

The following types of rivets may be required for the repair of the fuselage. Types of rivets are described in Section I.

Figure 50—Interior View of Wooden Fuselage

PART NO.	DESCRIPTION	
B1227-AD3	RIVET, BRAZIER HEAD	3/32-IN. DIA.
B1227-AD4	RIVET, BRAZIER HEAD	1/8 -IN. DIA.
B1227-AD5	RIVET, BRAZIER HEAD	5/32-IN. DIA.
AN426-AD3	RIVET, 100 C'SUNK	3/32-IN. DIA.
AN426-AD4	RIVET, 100 C'SUNK	1/8 -IN. DIA.
AN442-AD3	RIVET, FLAT HEAD	3/32-IN. DIA.
AN442-AD4	RIVET, FLAT HEAD	1/8 -IN. DIA.
AN442-AD5	RIVET, FLAT HEAD	5/32-IN. DIA.
AN442-AD6	RIVET, FLAT HEAD	3/16-IN. DIA.
LS-1127-4-2	RIVET, CHERRY BLIND BRAZIER	1/8 -IN. DIA

55. MISCELLANEOUS ALUMINUM REPAIR MATERIAL.

The following is a list of miscellaneous repair material that may be required for repair:

MATERIAL	SPECIFICATION
SHEET, 52S ALUMINUM	QQ-A-318
SHEET, 2S ALUMINUM	QQ-A-561
ROD, 2S ALUMINUM	QQ-A-411
PRIMER, ZINC CHROMATE	14080

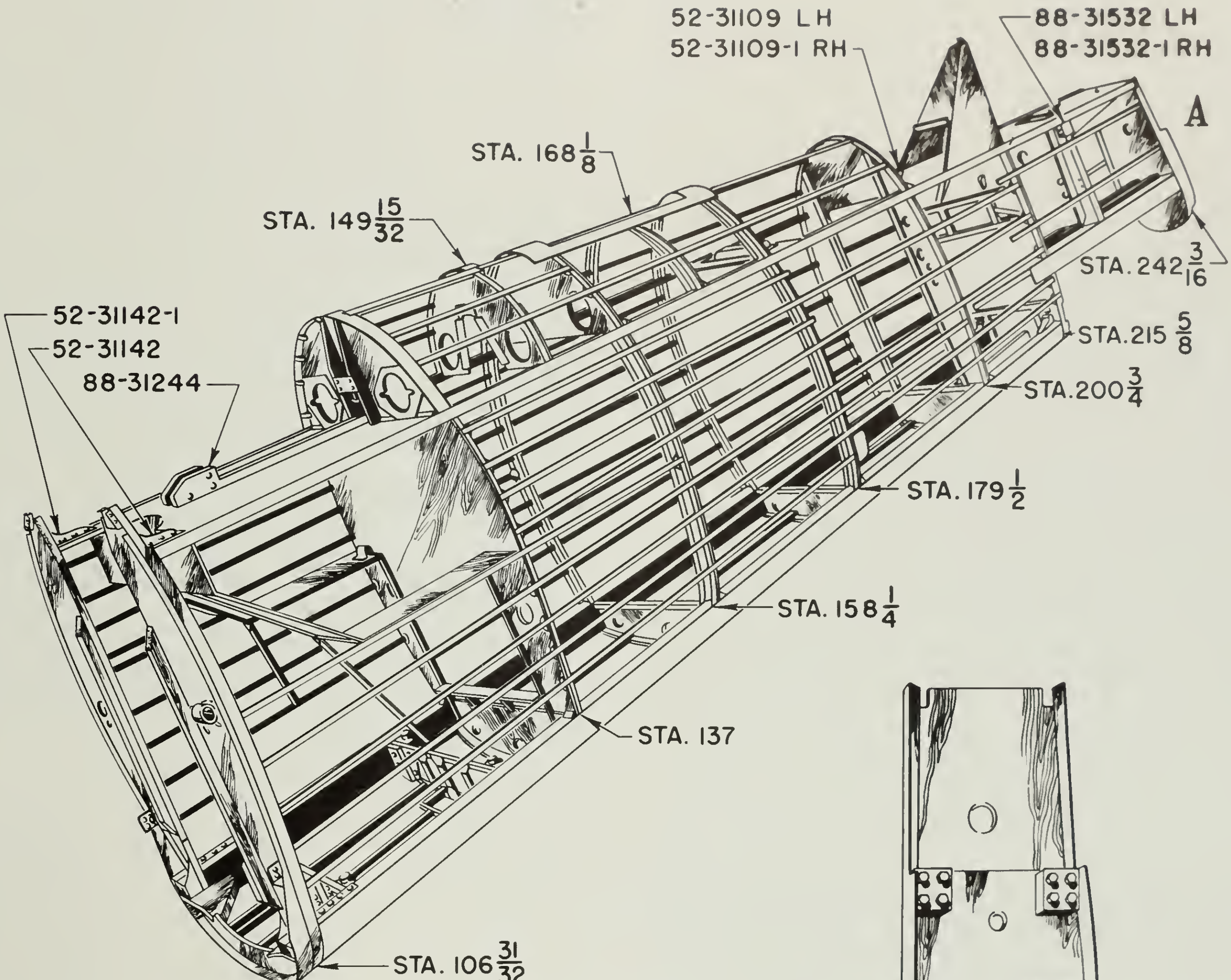
56. REPAIR TOOLS FOR ALUMINUM STRUCTURES.

For a complete list of sheet metal cutting, bending, forming, and riveting tools, see Section I.

57. WOODEN REAR FUSELAGE CONSTRUCTION.

On the later AT-6C Series Airplanes, the rear fuselage section from the rear cockpit to the tail is constructed of wood and the entire structure is assembled with casein glue. The wooden rear fuselage conforms in every respect in its general over-all dimensions, contours, and weight to the one of aluminum alloy construction used on the earlier AT-6C Series Airplanes and previous North American Aviation, Inc., trainer aircraft. Three-ply mahogany plywood (Spec. AN-NN-P511A) serves as the outside skin covering, bulkheads, frames, and shear webs. Solid spruce (Spec. AN-S-6) members serve as longerons, stringers, frame flanges, and bulkhead flanges *(see Figure 52)*. Maple compregwood is spliced to the spruce members only at points where it is necessary to absorb excessive loads. In several locations, 24ST aluminum alloy is used to reinforce fittings *(see Figure 51)*. In original construction, all the skin panels are scarf spliced, and the grain direction is laid at about 45 degrees to the vertical frames and bulkheads *(see Figure 53)*. To maintain the proper contour of the fuselage, there are eight frames spaced at approximately 21-inch intervals, some of the frames being complete bulkheads and others forming ring frames. The

Figure 51—Aluminum Alloy Reinforcement of Wooden Fittings



NOTES:

1. ALL WEB MEMBERS AND SKIN PANELS ARE MADE FROM 3-PLY MAHOGANY PLYWOOD CONFORMING TO AAF SPEC. AN-NN-P511A TYPE II
2. ALL STRINGERS AND FRAME FLANGES MADE FROM SPRUCE CONFORMING TO AAF SPEC. AN-S-6A
3. FOR ASSEMBLY CASEIN GLUE (AAF SPEC. C-G-456) IS USED.
4. ONLY FITTINGS ARE NOTED (THE ONLY REPLACEABLE ITEMS)

88-31515

(2 REQ)

DETAIL **A**

REF. DWG. 88-31106

Figure 52—Wooden Rear Fuselage Frame Assembly

stringer cutouts in the frames are reinforced with wooden gussets. The rectangular spruce stringers pass through the frames and are rigidly attached to the frames by the use of small, thin, tapered spruce wedges secured with casein glue. An intercostal located at the baggage compartment cutout is used to take direct bending loads, because of the shear transfer around the cutout. The two upper and two lower longerons support the entire fuselage bending loads, neglecting the effect of the continuous portions of the fuselage skin. The upper longeron is formed of rectangular spruce tapering toward the rear bulkhead. The lower longeron is made from rectangular spruce, several tricleats, and a plywood web, forming a closed triangular box section.

58. REPAIR WOOD MOISTURE CONTROL.

For obvious reasons, as much care and thoroughness must be taken with the repair of wooden aircraft parts as is taken with the original construction of the parts. For that reason, care should be taken to make certain that the moisture content of new or spliced parts is approximately the same as that in the original part. Wood gives off or takes on moisture from the surrounding air. Like many other hygroscopic materials, wood shrinks as it loses moisture and swells as it absorbs moisture. It is of the utmost importance that particular attention be paid to the moisture content of the part or parts to be repaired and the new wood to be used in the repair. If the part to be repaired has a moisture content of 10 percent and a piece of wood is spliced to it with a moisture content of 15 percent, when the finished assembly is out in service, both parts will lose or absorb moisture to the extent of finally having the same moisture content, depending upon the relative humidity and temperature. The sealer coats applied to the final wood assembly serve only to retard any increase or decrease in moisture content. Therefore, if the part having 15 percent moisture content goes down to 12 percent, it will shrink; and if the part having 10 percent goes up to 12 percent, it will swell, causing a strain in the glue joint. In general, if the wood is selected, prepared, and packaged in accordance with the applicable material specification for that particular type of wood, no consideration need be given to the moisture content of the wood, for it is necessarily the same as that used in original construction. If, however, there is any reason to doubt the correctness of the specified moisture content,

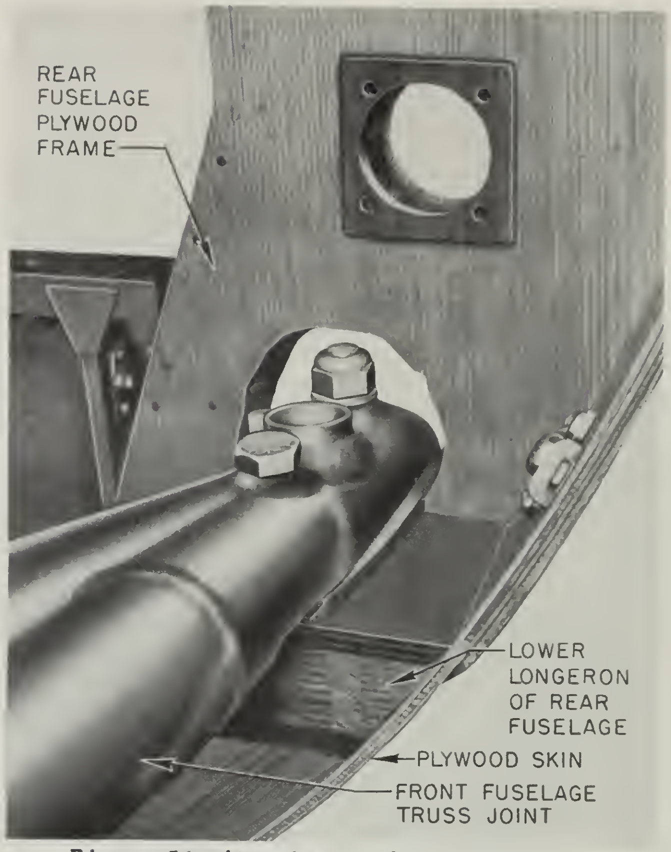


Figure 54—Attachment of Lower Steel Longeron to Rear Wooden Fuselage

Figure 53—Interior View of Fuselage Showing Skin Grain Direction

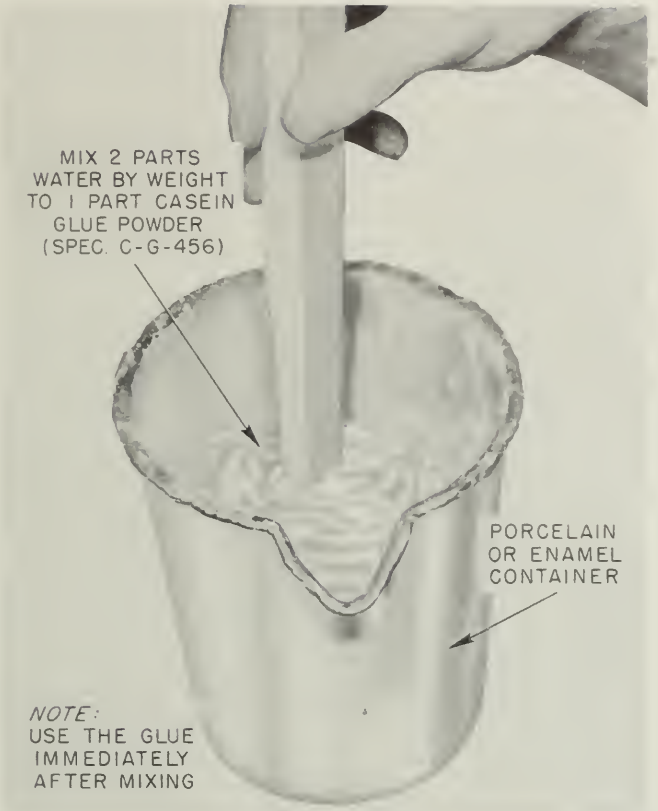


Figure 55—Mixing Casein Glue

because of storing the exposed wood for a considerable period, the moisture content of the wood should be checked and corrected before using the wood. The correct method of determining moisture content is to base the determination on oven-dry weight. The equipment necessary to determine moisture content consists of a drying oven and balance scale. The drying oven must be capable of maintaining a constant temperature of from 100° to 105°C (212° to 221°F). The balance scale must be accurate to within one half of one percent. The test specimen should be about one inch wide in the direction of the grain, about two inches long across the grain, and of the full depth of the material. The specimen should be taken far enough from the end to ensure that the moisture content has not been affected by end drying. All loose splinters must be removed and the test specimen weighed immediately after sawing, to avoid any moisture change resulting from exposure to air. The specimen should be placed in a drying oven and dried at 100° to 105°C (212° to 221°F) to constant weight. This will occur after about 24 hours. Immediately after removal from the oven, the specimen should be weighed. The formula for determining moisture content is as follows:

W = Original weight (before drying)
D = Oven-dry weight

The percentage of moisture content = $\frac{(W-D)}{D} \times 100$

If the moisture content of the wood is above the allowable limits specified in the applicable material specification for the particular type of wood, kiln-dry the wood-stock to the proper moisture content as outlined in Spec. AN-W-2 before the stock is used for repair. A variation of 2 percent in moisture content is the maximum permissible between the original structure and the patch or spliced replacement.

59. SUBSTITUTES FOR SPECIFIED WOODS.

In the repairs outlined in this Section, three-ply mahogany poplar core, plywood (Spec. AN-NN-P511A , and spruce (Spec. AN-S-6) are the only woods specified. In the event that these woods are not available, the following substitutes may be used. Any other substitutes are not acceptable and cannot be recommended. The following plywoods may be substituted for three-ply mahogany poplar core plywood and shall conform to AC Spec. AN-NN-P511A: (1) three-ply Douglas fir plywood, poplar core; (2) three-ply Douglas fir plywood, Douglas fir core; (3) three-ply red gum plywood, red gum core. The following woods may be substituted for spruce: (1) yellow poplar, Spec. AN-P-17; (2) western hemlock, Spec. AN-H-4; (3) noble fir,

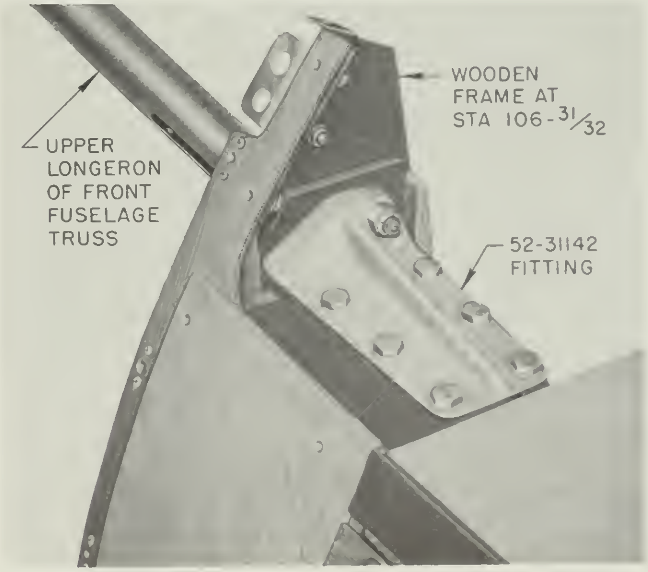


Figure 56—Attachment of Upper Steel Longeron to Rear Wooden Fuselage

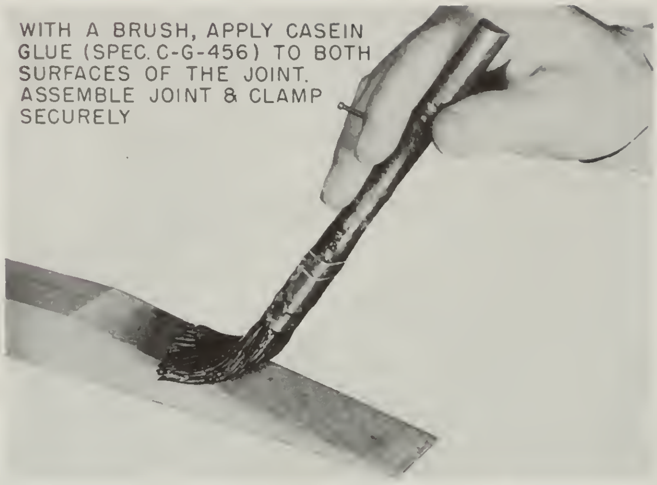


Figure 57—Applying Casein Glue With Brush

Spec. AN-F-6. In the case of a replacement only, no substitute for compregwood is permissible. Compregwood is defined as impregnated and compressed laminated maple having a density of from 80 to 85 pounds per cubic foot. Compregwood may be purchased from Formica Insulating Corp., Cincinnati, Ohio, or equivalent.

60. CASEIN GLUE FOR WOOD REPAIRS.

Casein glue is the only type glue used in the original construction of the wooden structures, and hence the only glue recommended for repairs. Any commercial casein glue conforming to Federal Spec. C-G-456 may be used. Casein, which is the chief protein constituent of milk, is the chief ingredient. The other ingredients are hydrated lime, sodium hydroxide, and water.

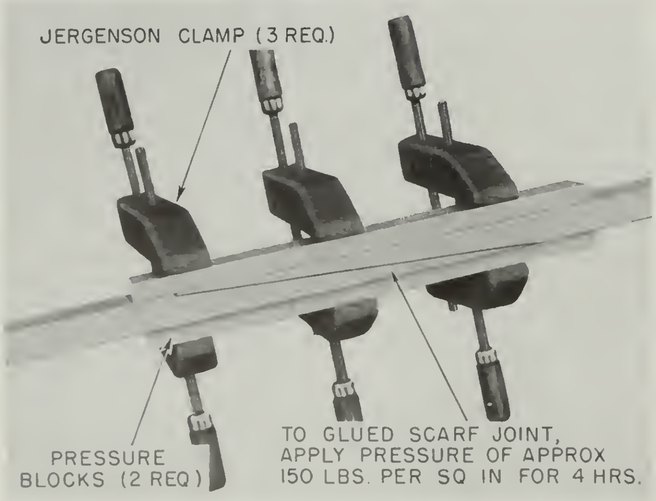


Figure 58—Correct Equalized Application of Gluing Pressure

This glue can be obtained in a powder form and mixed with water immediately prior to use. Casein glue is water-resistant but not waterproof, and is subject to mold growth.

61. MIXING CASEIN GLUE FOR WOOD REPAIRS.

All casein glue is obtained in the powdered form and mixed immediately prior to use. All proportions of glue and water used in mixing must be determined by weight and not by volume. The proportion of water and glue powder for mixing must conform to the directions which will be found on each container of certified glue, or which will be furnished by the manufacturer of the glue. In lieu of directions, a proportion of two parts of water by weight to one part of glue by weight is recommended for softwoods, such as spruce. For hardwoods, such as mahogany, use 1.8 parts of water by weight to one part glue powder by weight. The water should be at room temperature (21°C, 70°F). The use of a mechanical power-driven mixer is recommended to ensure full strength, the mixing to continue for approximately 5 minutes at a high speed of approximately 140 RPM. Stop the mixer when the glue thickens, and do not add water. Allow the mixture to stand for one-half hour for chemical digestion. Again operate the mixer for approximately 10 minutes at a slow speed of from 60 to 90 RPM, or for such additional time as required to dissolve all casein particles. The mixture must be of smooth consistency and free of air bubbles, foam, and lumps. If power-driven equipment is not available, hand methods may be substituted if the above procedure is followed as closely as possible. Ordinary room temperature and humidity are satisfactory for the mixing room. The glue must not be mixed in containers made of aluminum, brass, copper, or zinc. Small containers made of porcelain or enamelware are preferred *(see Figure 55)*; larger containers or mixing bowls shall be of iron or steel. All containers and brushes must be kept thoroughly clean at all times and must contain none of the remaining glue of the preceding batch. The glue must be used within 4 hours after mixing, for the glue jells after a longer period and cannot be used. Do not heat the glue or the wood. The mixing and application of casein glue is strictly a cold process.

62. ASSEMBLING GLUED JOINTS OF WOOD.

Apply a sufficient quantity of glue to all joints to penetrate into the pores of the wood, and coat all parts of the joint. The glue and the wood shall be at room temperature, and the surfaces of the wood for gluing shall be smooth and cleaned of all greases, varnish, or other coating materials. Tooth-planing or scratching is not recommended. However, when gluing compregwood, the surface should be roughened before the application of glue. The glue shall not be spread thin or brushed out in thin layers. A spread of approximately 1-1/2 ounces of wet glue per square foot of single glue line is satisfactory. Where practicable and where repair conditions will permit, the glue should be applied to both surfaces. A mechanical spreader is preferred; but as most repairs will cover only small areas, a brush may be used *(see Figure 57)*. Brushes must be of good quality to prevent loose or broken bristles becoming mixed with the film of glue. In most cases, clamps should be applied to all joints immediately after the glue is applied, and should be adjusted to give a uniform pressure on all parts of the joint. Any type of clamp may be used that will give satisfactory results and that will exert a pressure of approximately 150 lbs./sq.in. for softwood and 200 lbs./sq.in. for hardwood *(see Figure 58)*. The clamps must not be removed from the joints within 4 hours after clamping, and a longer period is preferred and required where joints cover an extended area or where it is necessary to use large quantities of glue in proportion to the volume of wood in the member. Handwork on glued parts may be done as soon as they are removed from the clamps, but machining should not be attempted until after 12 hours. When gluing compregwood or wood of extremely fine-grained nature, it is often desirable to coat both surfaces with an even coat of glue and allow it to set until it reaches a state of tackiness. Apply a final coat of glue and clamp the joint immediately.

63. APPLICATION OF WOOD-GLUING PRESSURE WITH CLAMPS.

In most cases, pressure for gluing may be applied with clamps, but care should be taken to provide uniform pressure *(see Figure 58)*. The importance of properly applying gluing pressure cannot be stressed too strongly. Uniform pressure may be provided by first applying pressure blocks to both sides of the joint and then clamping the area *(see Figure 58)*. Use a sufficient number of clamps to provide a uniform pressure; otherwise a poor joint will result *(see Figure 59)*. Various types of clamps are available, and each has its own particular advantage. The most commonly used wood clamp is the Jorgensen clamp *(see Figure 61)*. The Jorgensen clamp has two wooden jaws equipped with two hand screws for exerting pressure. Another clamp, the bar clamp, is made up of a bar about 2 inches wide and from 18 inches long to 6 feet long, depending upon the type *(see Figure 61)*. The "C" clamp is available in various sizes and is probably the most useful type of clamp for repair work *(see Figure 61)*. Any type of clamp may be used that will give satisfactory results and that will exert a pressure of approximately 150

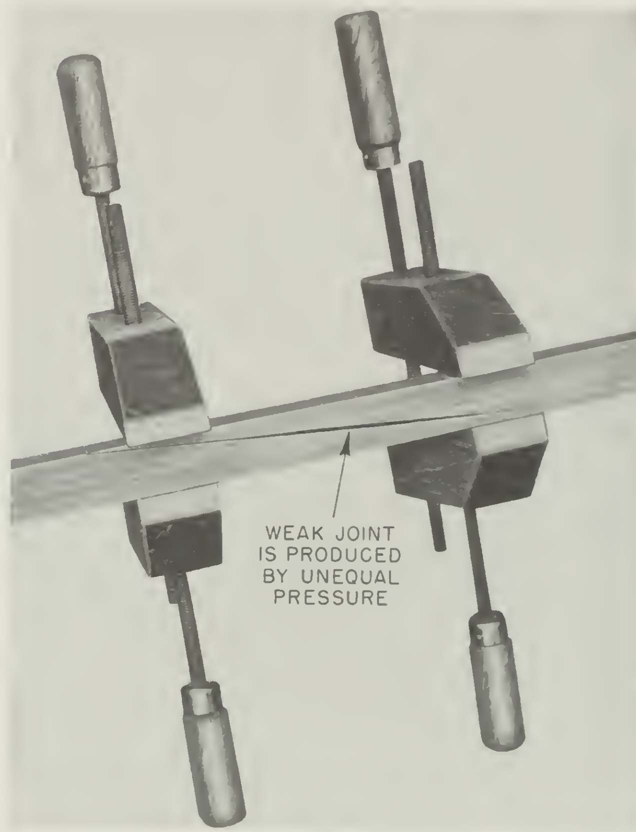


Figure 59—Incorrect Concentrated Application of Gluing Pressure

lbs./sq.in. for softwood and 200 lbs./sq.in. for hardwood. The clamps must not be removed from the joints within 4 hours after clamping, and a longer period is preferred and required where joints cover an extended area.

64. APPLICATION OF WOOD-GLUING PRESSURE WITH NAILING STRIPS.

In assembling the skin to the interior structure, and in other instances where the application of clamps is impossible, pressure required for gluing can be obtained by temporarily nailing strips of wood over the glued area. These nailing strips are generally made of basswood or soft pine; however, spruce or any other available wood can be used. The nailing strips are cut 1/4-inch thick and 3/8-inch to 1/2-inch wide depending upon the use. The size of nails used varies according to the thickness and width of the material to be glued. For securing 1/16-inch to 3/32-inch thick plywood skin to stringers while the glue is hardening, a spacing of 1-1/2 inches apart, using a 5/8-inch No. 20 gage nail, is recommended *(see Figure 60)*. Sufficient nailing strips must be used to cover entirely the scarf joint area. For example, if the width of the scarf joint is 1-1/2 inches, three 1/2-inch wide nailing strips must be used to provide pressure. Place the nailing strips side by side and alternate the nail positions in the adjacent strips. The nailing strips should be secured first to the center of the scarf joint length and then the nailing should progress outward from the center. Drive the nails down as tightly as possible, in order to provide adequate gluing pressure. After the nailing strips are secured, the glued

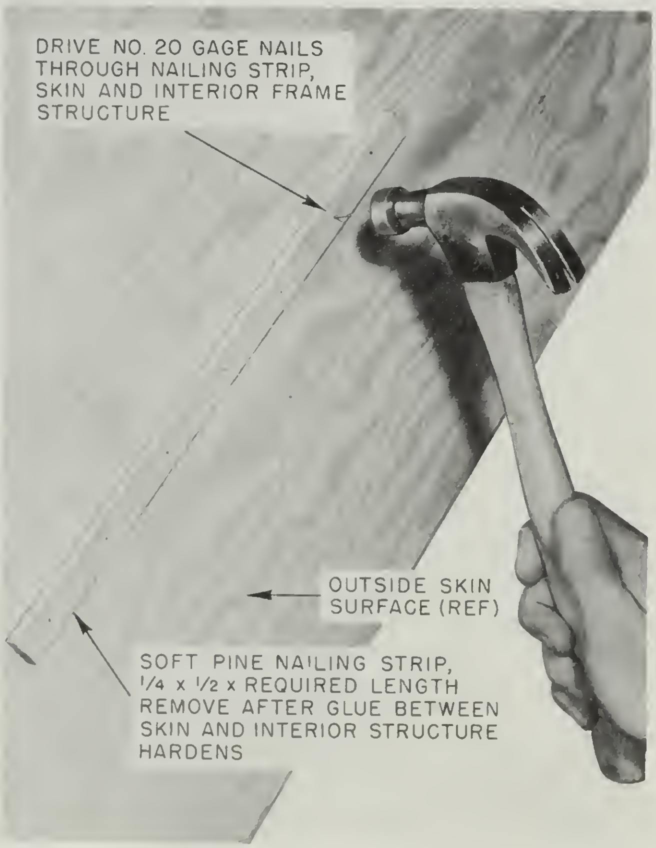


Figure 60—Using Nailing Strips to Apply Gluing Pressure to Skin

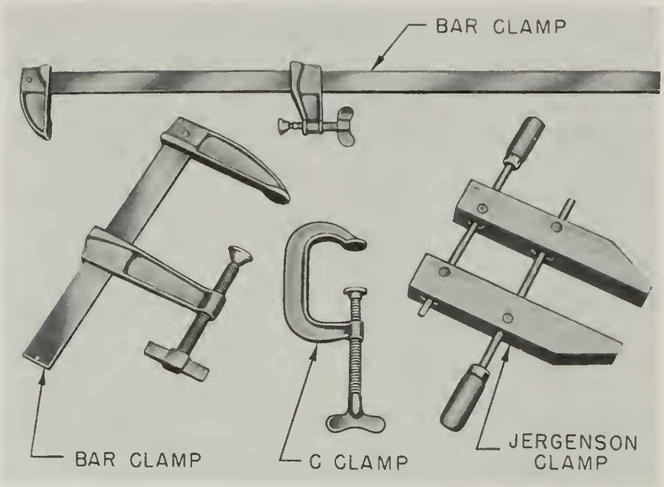


Figure 61—Types of Clamps for Wood

skin portion must be free from wrinkles, bulges, and joint gaps. To prevent glue from seeping from the joint and sticking to the nailing strip, waxed paper may be placed under the nailing strips. After the glue has hardened, remove the nails and the nailing strips.

65. WOODEN REAR FUSELAGE STRINGERS.

The wooden rear fuselage stringers are formed of rectangular spruce having a thickness of 1/2-inch and a width of 5/8-inch. Damaged stringers may be repaired by splicing in a new stringer section as shown *(see Figure 62)*. If the damage is extensive, at each side of the damage, diagonally cut out the stringer section with a small keyhole saw *(see Figure 76)*. Make these cuts at a ten-to-one taper which amounts to a 5-inch taper on this particular type of stringer *(see Figure 62)*. Plane the stringer material down to the skin. With a chisel, pare off any small remaining portions of the damaged stringer adjacent to the scarf which may be inaccessible for the manipulation of the wood plane *(see Figure 63)*. During these cutting operations, take particular care to prevent damage to the skin. Cut a replacement stringer section of spruce (Spec. AN-S-6) slightly longer than the removed portion of the original stringer and having cross-sectional dimensions of 1/2- by 5/8-inch. Scarf-cut the ends of the replacement stringer to match the cuts on the original stringer. Carefully plane down the saw-cut ends of the scarf cuts on the original stringer and the replacement *(see Figure 65)* so that they are perfectly smooth when measured with a straightedge *(see Figure 67)*, and so that they form a perfect fit when assembled. Do not use sandpaper

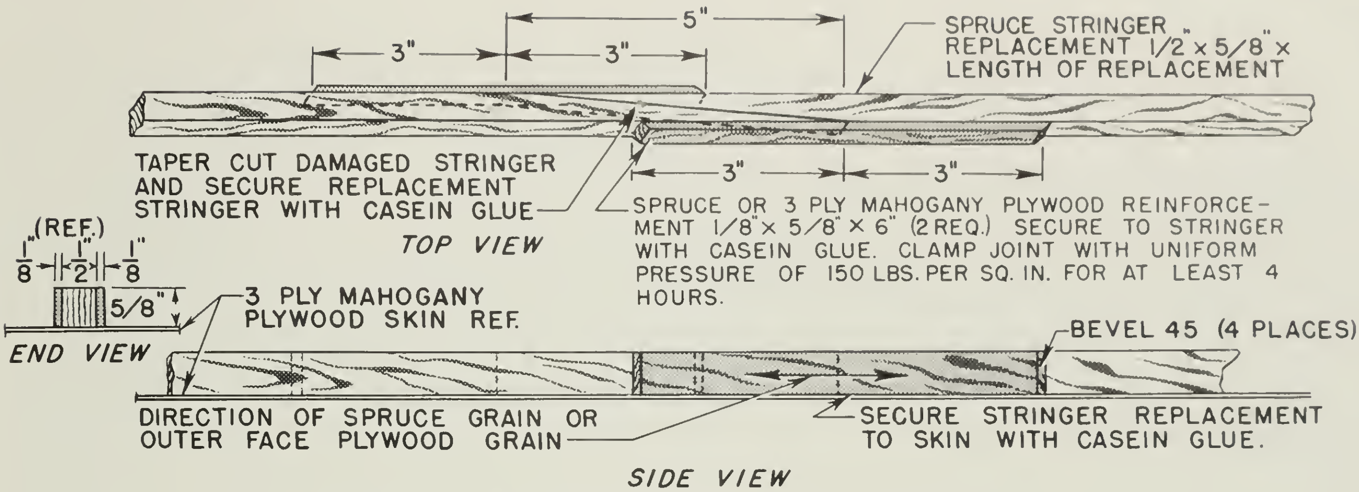


Figure 62—Wooden Rear Fuselage Stringer Splice

on surfaces which are to be glued, since the wood dust particles prevent proper gluing. To serve as the reinforcements for each of the scarf joints, cut two spruce sheets (Spec. AN-S-6) or two sheets of three-ply mahogany poplar core plywood (Spec. AN-NN-P511A), each 1/8-inch thick 5/8-inch wide and 6 inches in length *(see Figure 62)*. Clean all wood chips from the surfaces to be glued. With a brush, apply one coat of casein glue (Spec. C-G-456) to all joint surfaces and to the skin and replacement stringer contact surfaces. Assemble the scarf joint as shown *(see Figure 62)* and immediately apply a series of clamps to the joint area to provide a uniform pressure of approximately 150 lbs./sq.in. To provide gluing pressure along the skin for the stringer replacement, cut a strip of soft pine 1/4-inch thick and 1/2-inch wide and having a length equal to the length of the stringer replacement. On the outside skin surface, nail the soft-pine strip to the stringer through the skin, using 5/8-inch long No. 20 gage nails at 1-1/2 inch spacing *(see Figure 60)*. Allow the glue to remain under pressure for at least 4 hours; then remove the clamps and nailing strip. Carefully scrape off all excess glue around the joints. To the entire added wood material, apply one coat of sealer (AAF Spec. 14113) and allow to dry for at least one hour. Apply a second coat of sealer (AAF Spec. 14113) to all added material except mahogany if used, and to the end-grain surfaces of the 1/8-inch reinforcement. To mahogany and the end-grain surfaces of the 1/8-inch reinforcement and to the nail holes in the outside skin, rub on one coat of unthinned filler (NAA Formula No. 1398 or equivalent), allowing the filler to remain on the surface from 2 to 5 minutes; then remove the excess filler with excelsior. Allow filler to dry for at least 2 hours; then to all added material except the end-grain surfaces, apply one finish coat of either cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98) as required to match adjacent surfaces. Apply finish tape to the end-grain

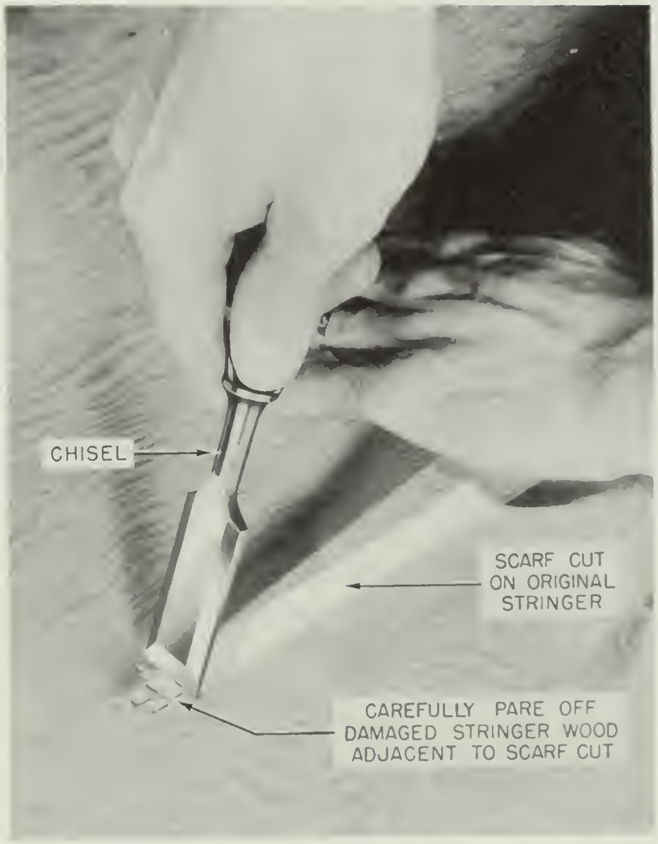


Figure 63—Removing Portion of Damaged Wood Stringer Adjacent to Scarf Cut

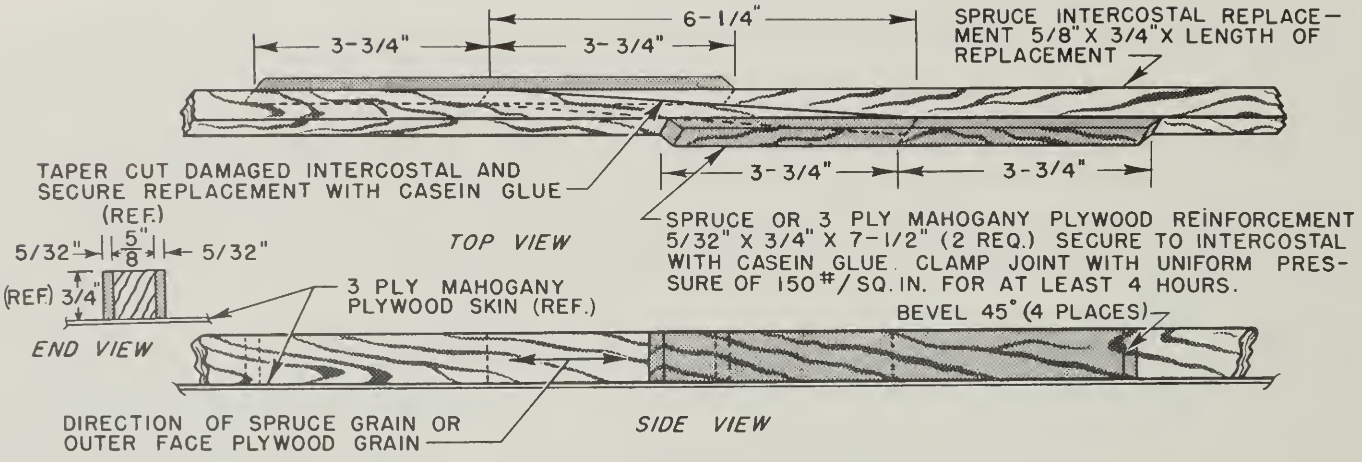


Figure 64—Baggage Compartment Wooden Intercostal Splice

surface as outlined in the paragraph applicable to finishes for wood.

66. WOODEN REAR FUSELAGE BAGGAGE COMPARTMENT INTERCOSTAL.

The NA 88-31108-29 intercostal located on the left side of the rear fuselage below the baggage compartment may be repaired by splicing in a new intercostal angle *(see Figure 64)*. The intercostal extends 51 inches aft from the fuselage monocoque joining surface and is formed of rectangular spruce having a thickness of 5/8-inch and a width of 3/4-inch. If the damage is extensive, make a diagonal cut with a keyhole saw on one side of the damage so that damaged section is cut free from the longest length of undamaged intercostal. Make this cut at a ten-to-one taper, which amounts to a 6-1/4 inch taper on this particular type of intercostal *(see Figure 64)*. Carefully plane off the shorter damaged length of the intercostal until the skin is bared. With a chisel, pare off any small remaining portions of the damaged intercostal adjacent to the scarf which may be inaccessible for the manipulation of the wood plane. During these cutting operations, take particular care to prevent damage to the skin. Cut a replacement intercostal section of spruce (Spec. AN-S-6) slightly longer than the removed portion of the original intercostal and having cross-sectional dimensions of 1/2- by 5/8-inch. Scarf-cut the end of the replacement intercostal to match the cut end of the original intercostal. Carefully plane down the ends of the scarf cuts *(see Figure 65)* so that they are perfectly smooth when measured with a straightedge, and so that they form a perfect fit when assembled *(see Figure 67)*. Do not use sandpaper on surfaces which are to be glued, since the wood dust particles prevent proper gluing. Cut two spruce sheets (Spec. AN-S-6) or two sheets of three-ply mahogany poplar core plywood (Spec. AN-NN-P511A) each 5/32-inch thick, 3/4-inch wide, and 7-1/2 inches in length *(see Figure 64)*. Clean all wood chips from the surfaces to be glued. With a brush, apply one coat of casein glue (Spec. C-G-456) to all joint surfaces and to the skin and replacement intercostal contact surfaces. Assemble the scarf joint as shown *(see Figure 64)* and immediately apply a series of clamps to the joint area to provide a uniform pressure

Figure 65--Smoothing Scarf Cut With a Wood Plane

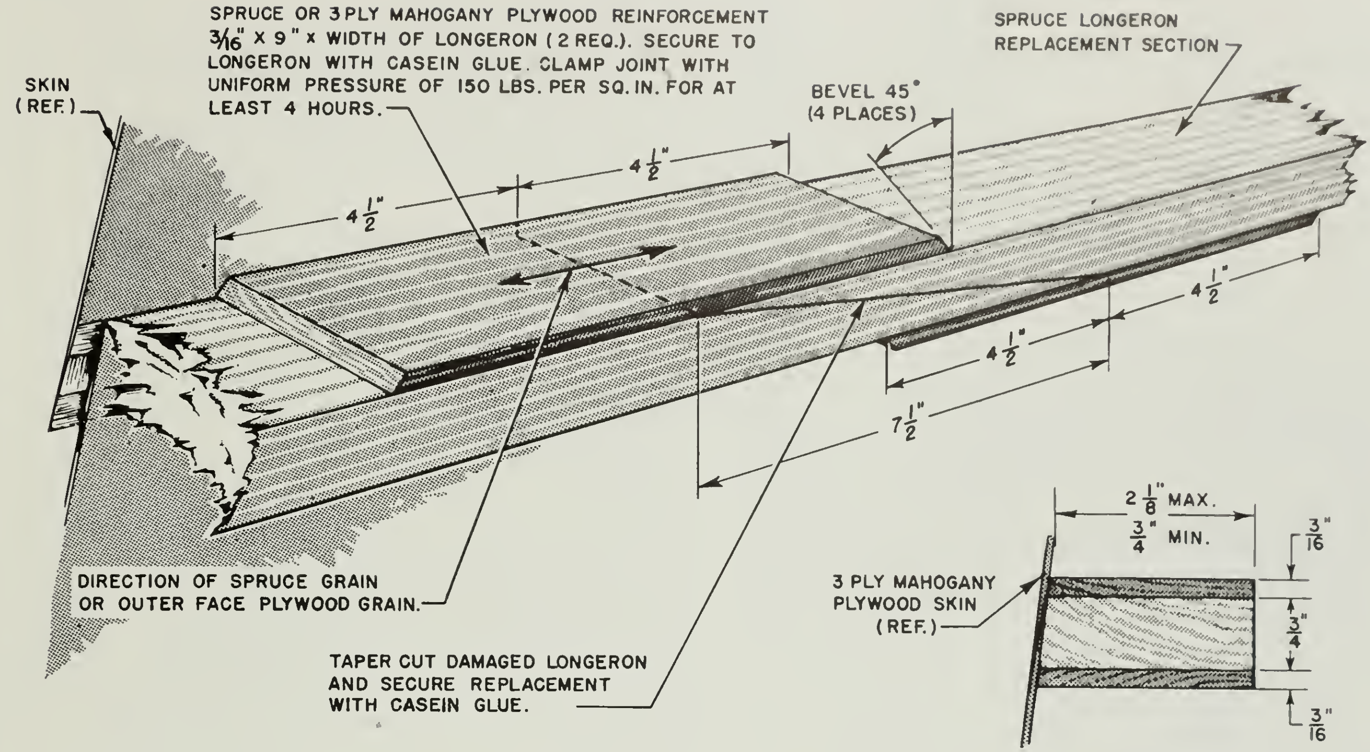


Figure 66—Wooden Rear Fuselage Upper Longeron Splice

of approximately 150 lbs./sq.in. *(see Figure 58)*. To provide gluing pressure along the skin for the intercostal replacement, cut a strip of soft pine 1/4-inch thick and 1/2-inch wide and having a length equal to the length of the intercostal replacement. On the outside skin surface, nail the soft-pine strip to the stringer through the skin, using 5/8-inch long No. 20 gage nails at 1-1/2 inch spacing *(see Figure 60)*. Allow the glue to remain under pressure for at least 4 hours; then remove the clamps and nailing strip. Carefully scrape off all excess glue around the joints. To the entire added wood material, apply one coat of sealer (AAF Spec. 14113) and allow to dry for at least one hour. Apply a second coat of sealer (AAF Spec. 14113) to all added material except to mahogany if used, and to the end-grain surfaces of the 5/32-inch reinforcements. To mahogany and the end-grain surfaces of the 5/32 inch reinforcements and to the nail holes in the outside skin, rub on one coat of unthinned filler (NAA Formula No. 1398 or equivalent), allowing the filler to remain on the surface from 2 to 5 minutes; then remove the excess filler with excelsior. Allow the filler to dry for at least 2 hours; then to all added material except the end-grain surfaces, apply one finish coat of either cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98) as required to match adjacent surfaces. Apply finish tape to the exposed end-grain surfaces as outlined in the paragraph applicable to finishes for wood.

67. WOODEN REAR FUSELAGE UPPER LONGERONS.

The wooden rear fuselage upper longeron (NA 88-31552) which extends 120 inches aft of the

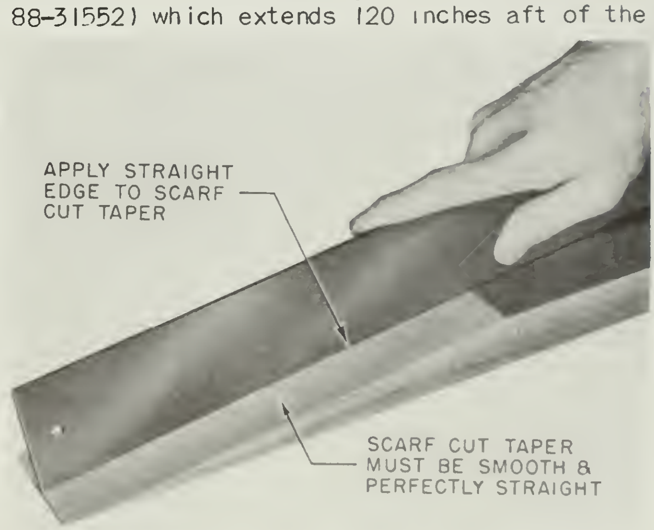


Figure 67—Checking Scarf Cut With Straightedge

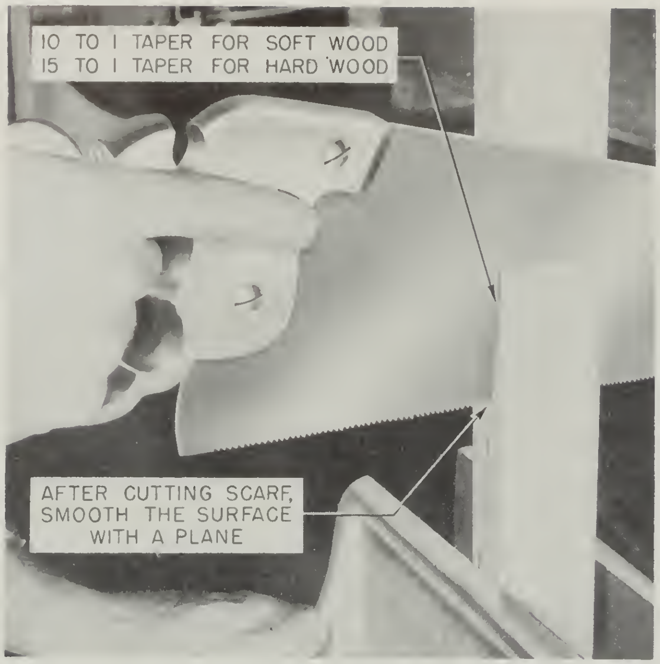


Figure 68—Cutting Wood Scarf Joint

fuselage joining surface on the left and the right side of the fuselage is formed of rectangular spruce having a thickness of 3/4-inch and a width which tapers from 2-1/8 inches at the forward position to a 3/4-inch width at the aft location. At the forward fuselage joining surface, the longeron is spliced to a compregwood section. The complete replacement of a damaged longeron is not recommended, inasmuch as splicing and partial replacement may be accomplished with far less work. No repairs to the upper longeron are permissible within the first 40 inches aft of the fuselage joining surfaces. If damage occurs in this area, replace the forward longeron section and locate the splice aft of the fuselage joining surface. If the damage is extensive, cut out the damaged material with a keyhole saw. Make the cut at a ten-to-one taper, which amounts to a 7-1/2 inch taper on this particular type of longeron *(see Figure 66)*. Carefully plane off the damaged length of the longeron until the skin is bared. With a chisel, pare off any small remaining portions of the damaged longeron adjacent to the scarf which may be inaccessible for the manipulation of the wood plane. During these cutting operations, take particular care to prevent damage to the skin. Cut a replacement longeron section of spruce (Spec. AN-S-6) slightly longer than the removed portion of the original longeron and having maximum cross-sectional dimensions of 3/4- x 2-1/8 inches *(see Figure 66)*. Scarf-

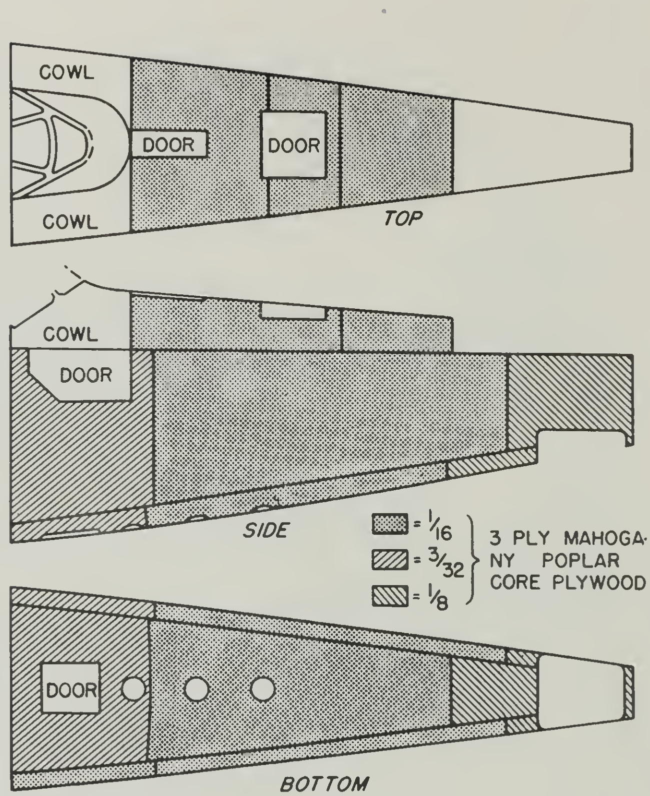


Figure 69—Wooden Rear Fuselage Skin Arrangement

cut the ends of the replacement longeron to match the cut end of the original longeron. Carefully plane down the ends of the scarf cuts so that they are perfectly smooth when measured with a straightedge *(see Figure 67)*, and so that they form a perfect fit when assembled. Do not use sandpaper on surfaces which are to be glued, since the wood dust particles prevent proper gluing. Cut two spruce sheets (Spec. AN-S-6) or two sheets of three-ply mahogany poplar core plywood (Spec. AN-NN-P511A), each 3/16-inch thick, 9 inches long, and having a width equal to the width of the longeron at the affected area *(see Figure 66)*. Clean all wood chips from the surfaces to be glued. With a brush, apply one coat of casein glue (Spec. C-G-456) to all joint surfaces and to the skin and replacement longeron contact surfaces. Assemble the scarf joint as shown *(see Figure 66)*, and immediately apply a series of clamps to the joint area to provide a uniform pressure of approximately 150 lbs./sq. in. *(see Figure 58)*. To provide gluing pressure along the skin for the longeron replacement section, cut a strip of soft pine 1/4-inch thick and 1/2-inch wide and having a length equal to the length of the longeron replacement. On the outside skin surface, nail

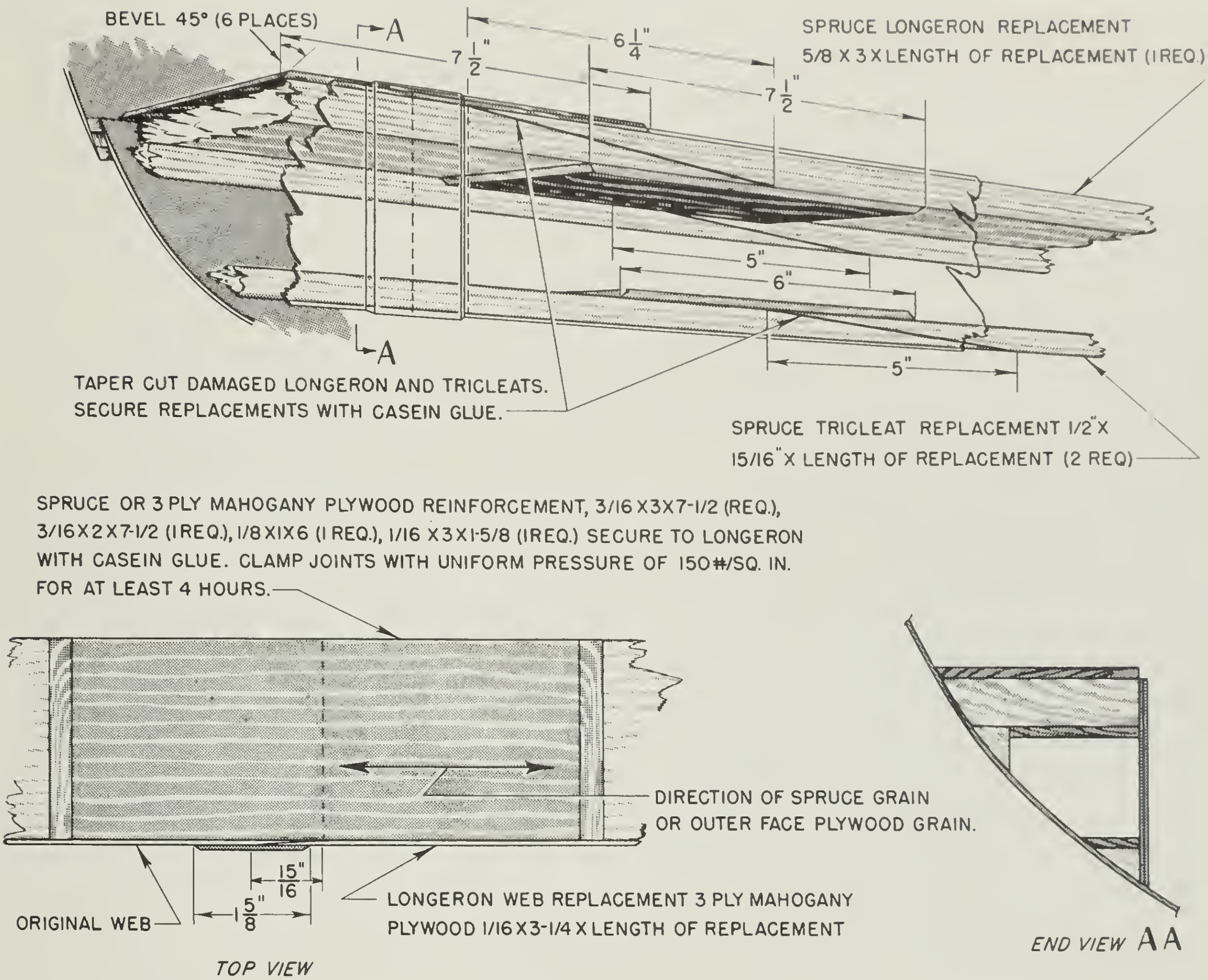


Figure 70—Wooden Rear Fuselage Lower Longeron Splice

the soft-pine strip to the longeron through the skin, using 5/8-inch long No. 20 gage nails at one-inch spacing. Allow the glue to remain under pressure for at least 4 hours; then remove the clamps and the nailing strip. Carefully scrape off all excess glue around the joints. To the entire added wood material, apply one coat of sealer (AAF Spec. 14113), and allow to dry for at least one hour. Apply a second coat of sealer (AAF Spec. 14113) to all added material except mahogany if used, and to the end-grain surfaces of the 3/16-inch reinforcements. To mahogany and the end-grain surfaces of the 3/16-inch reinforcements and to the nail holes in the outside skin, apply one coat of unthinned filler (NAA Formula No. 1398 or equivalent), allowing the filler to remain on the surface from 2 to 5 minutes; then remove the excess filler with excelsior. Allow the filler to dry for at least 2 hours; then to all added material except the end-grain surfaces, apply one finish coat of either cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98) as required to match adjacent surfaces. Tape the exposed end-grain surfaces of the reinforcements as outlined in the paragraph applicable to finishes.

68. WOODEN REAR FUSELAGE LOWER LONGERONS.

The wooden rear fuselage lower longerons (NA 88-31121), which extend 110 inches aft of the fuselage joining surface on the left and the right side of the fuselage, are each formed of a main member of rectangular spruce having a thickness of 5/8-inch and a width of approximately 3 inches. This main spruce member is supported along the entire length by two spruce tricleat sections and a plywood web *(see Figure 70)*. At both the front and rear ends of the longeron, the main spruce member of the longeron is spliced to a short compregwood section for greater strength. The complete replacement of a damaged longeron is not recommended, but splicing and partial replacement may be accomplished. No repairs to the lower longeron are permissible at the compregwood ends of the longerons. If damage occurs in this area, replace the affected longeron section, and locate the splice beyond the compregwood area. At the location where the splice is to be located, vertically cut the longeron web and

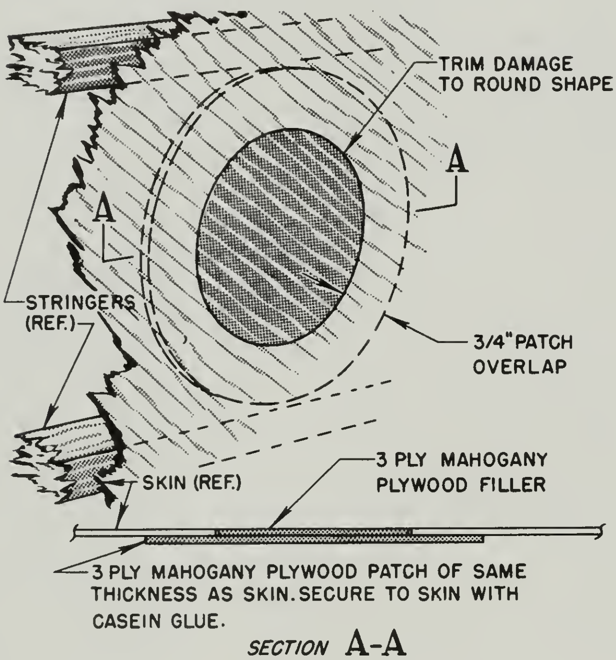


Figure 71—Flush Patch for Plywood Skin Holes Less Than 2 Inches Wide

remove the affected web section as required *(see Figure 70)*. If the damage is extensive, cut out the damaged longeron material with a keyhole saw. On the main spruce member of the longeron, make the cut at a ten-to-one taper, which amounts to a 7-1/2 inch taper on this particular section *(see Figure 70)*. If the two spruce tricleats are damaged, taper cut out the damaged material as noted *(see Figure 70)*. Carefully plane off the length of the damaged longeron and tricleats until the skin is bared. With a chisel, pare off any small remaining portions of the damaged material adjacent to the scarf cuts which may be inaccessible for the manipulation of the wood plane. During these cutting operations, take particular care to prevent damage to the skin. Cut replacement tricleats and a new longeron section of spruce (Spec. AN-S-6) to match the removed sections. Scarf-cut the ends of the replacement members to match the cut ends of the original longeron sections. Carefully plane down the ends of the scarf cuts *(see Figure 65)* so that they are perfectly smooth when measured with a straightedge *(see Figure 67)*, and so that they form a perfect fit when assembled. To serve as reinforcements, cut the four pieces of spruce or four pieces of three-ply mahogany poplar core plywood to the sizes noted *(see Figure 70)*. Do not use sandpaper on surfaces which are to be glued, since the wood dust particles prevent proper gluing. Clean all wood chips from the surfaces to be glued. With a brush, apply one coat of casein glue (Spec. C-G-456) to all longeron and tricleat joint surfaces and to the skin and replacement longeron contact surfaces. Assemble the scarf joints as shown *(see Figure 70)*, and immediately apply a series of clamps to the joint area to provide a uniform pressure of approximately 150 lbs./sq.in. To provide gluing pressure along the skin for the replacement sections, cut strips of soft pine 1/4-inch thick and 1/2-inch wide and having a length equal to the length of the replacement. On the outside skin surface, nail the pine strips to the replacement sections through the skin, using 5/8-inch long No. 20 gage nails at 1-1/2 inch spacing. Use a sufficient number of nailing strips to cover completely the glued area. Allow the glue to remain under pressure for at least 4 hours; then remove the clamps and nailing strips. Carefully scrape off all excess glue around the joints. To the entire added wood material except the area to be glued to the longeron web, apply two coats of sealer (AAF Spec. 14113), allowing each coat to dry for at least one hour. To all end-grain surfaces, apply one coat of unthinned filler (NAA Formula No. 1398 or equivalent), allowing the filler to remain on the surface from 2 to 5 minutes; then remove the excess filler with excelsior. To serve as the longeron web replacement, cut a sheet of three-ply mahogany plywood 1/16-inch thick, 3-1/4 inches wide, and having a length equal to the length of the replacement. With a spokeshave, scarf the ends of the longeron, plywood web *(see Figure 73)*, original web, and replacement web at a 15/16-inch taper so that they form a perfect fit when assembled. Apply two coats of sealer (AAF Spec. 14113) to the inside of the longeron web except the areas to be glued. Assemble the longeron web splice and reinforcement with casein glue (Spec. C-G-456), and apply gluing pressure for at least 4 hours. To the web replacement, apply two coats of sealer (AAF Spec. 14113) allowing each coat to dry for at least one hour. To all outside surfaces except the end-grain surfaces of the web reinforcement, apply one finish coat of either cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98) as required to match adjacent surfaces. Tape the end-grain surfaces of the web reinforcement as outlined in the paragraphs applicable to finishes.

69. FLUSH PATCH FOR PLYWOOD SKIN HOLES LESS THAN 2 INCHES WIDE.

Wherever possible, the flush patch should be used for small holes in the skin. However, where the application of this flush patch is not feasible because of stringers immediately adjacent to the skin hole, the external patch or skin section replacement may be used. The procedure for the application of an inner patch to the fuselage skin is as follows: With a small hole saw, cut out the damage to a circular shape so that the removed skin includes all cracks and broken edges. The trimmed skin hole should not exceed 2 inches. To serve as the patch, cut a circular piece of three-ply mahogany poplar core plywood of the same thickness as the skin and large enough to lap the skin hole by 3/4-inch all around *(see Figure 71)*. To serve as the filler, cut another circular piece of three-ply mahogany poplar core plywood of the same thickness as the skin and of the same shape as the removed material. Apply one coat of casein glue to the contact surfaces of the patch and filler and assemble as shown *(see Figure 71)*. To provide gluing pressure, apply a short strip of soft pine over the filler and temporarily nail the filler to the patch with No. 20 gage nails passed through the pine strip. With a small wood plane, carefully plane off a very thin layer of wood on the inside of the skin around the hole where the patch is to be applied. To the end-grain surfaces of the skin hole, filler, and patch, apply one coat of sealer (AAF Spec. 14113). Allow to dry for one hour; then apply one coat of unthinned filler (NAA Formula No. 1398 or equivalent , allowing the filler to remain on the surface from 2 to 5 minutes. Then remove the excess filler with excelsior. Apply one coat of casein glue to the contact surfaces of the patch and the skin, and assemble as shown *(see Figure 71)*. To provide gluing pressure, apply a short strip of soft pine over the patch and temporarily nail the patch to the skin in several places with No. 20 gage nails passed through the pine strip. Allow the glue to dry for at least 4 hours; then pull out the nails and remove the nailing strips. To the accessible patch area on the inside of the fuselage, apply one coat of sealer (AAF Spec. 14113), allow to dry for one hour, and then apply one coat of either cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98) as required to match adjacent surfaces. To the outside surface of the filler, apply two coats of sealer, allowing a drying time of one hour between coats. Then apply one coat of surfacer (AAF Spec. 14115), allow to dry, and sand off. Lastly, apply one spray coat of aircraft enamel (Spec. 3-98) pigmented with 16 ounces of aluminum paste (Spec. AN-TT-A-461) per gallon of unthinned enamel.

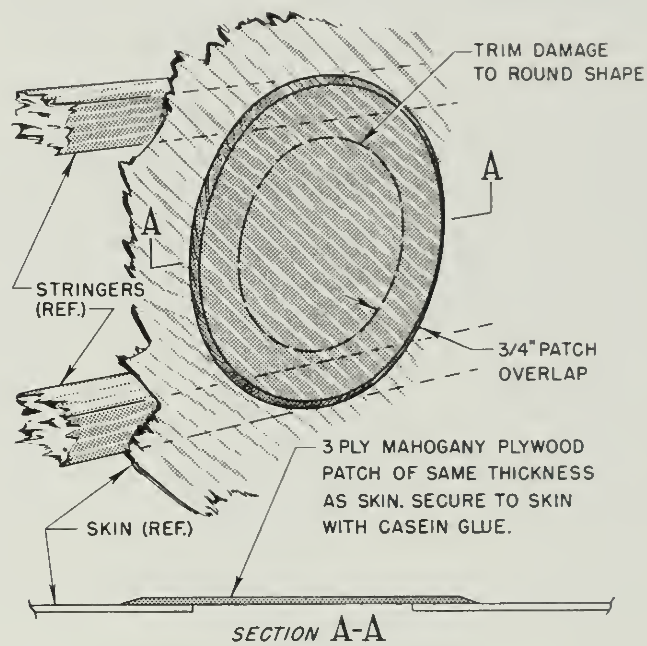


Figure 72—External Patch for Plywood Skin Holes Less Than 2 Inches Wide

70. EXTERNAL PATCH FOR PLYWOOD SKIN HOLES LESS THAN 2 INCHES WIDE.

The external patch should be used only in an emergency where the application of a small inner-flush patch or a section replacement is impossible. With a small hole saw, cut out the damage to a circular shape so that the removed skin includes all cracks and broken edges. The trimmed hole should not exceed 2 inches. To serve as the patch, cut a circular piece of three-ply mahogany poplar core plywood (Spec. AN-NN-P-511A) of the same thickness as the skin

Figure 73—Scarfing Plywood Skin With Spokeshave

and large enough to lap the skin hole by 3/4-inch all around *(see Figure 72)*. Bevel the edges of the patch. With small wood plane, carefully plane off the finish on the outside of the skin around the hole where the patch is to be applied. Apply one coat of casein glue (Spec. C-G-456) to the contact surfaces of the patch and the skin, and apply the patch. To provide gluing pressure, apply a short strip of soft pine over the patch and temporarily nail the patch to the skin in several places with No. 20 gage nails passed through the pine strip. Allow the glue to remain under pressure for at least 4 hours; then remove the nails and nailing strip. Carefully scrape off all excess glue around the joint. To the inside and the outside of the patch, apply one coat of sealer (AAF Spec. 14113) and allow to dry for at lease one hour. Apply a second coat of sealer to the inside and the outside of the patch except to the end-grain surfaces of the skin hole and patch. To the end-grain surfaces of the skin hole and patch, apply one coat of unthinned filler (NAA Formula No. 1398 or equivalent), allowing the filler to remain on the surface from 2 to 5 minutes; then remove the excess filler with excelsior. Allow the filler to dry for at least 2 hours; then to the outside of the patch, apply one coat of surfacer (AAF Spec. 14115) and allow to dry. To the inside of the patch, apply one coat of either cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98) as required to match adjacent surfaces. To the outside of the patch, apply one coat of clear aircraft enamel (Spec. 3-98) pigmented with 16 ounces of aluminum paste (Spec. AN-TT-A-461) per gallon of unthinned enamel. This completes the repair procedure.

71. REPLACING DAMAGED PLYWOOD SKIN SECTION.

If a hole in plywood skin exceeds 2 inches in width, cut out the entire affected plywood area in a longitudinal direction at the nearest stringer on each side of the damage and as far as required in a transverse direction between frames *(see Figure 74)*. Use a fine keyhole saw to remove the main portion of the skin, and a chisel for the removal of small areas inaccessible for the manipulation of the saw. Use great care to cut the edges of the plywood straight and square. After the skin is thus trimmed, scarf the skin edges at a 15-to-1 taper vith a spokeshave or a drawknife *(see Figure 73)*. This scarf taper amounts to a 15/16-inch scarf on 1/16-inch skin, a 1-13/32 inch taper on 3/32-inch skin, and a 1-7/8 inch taper on 1/8-inch skin. *(See Figure 74.)* The scarf cuts must be smooth and perfectly straight when measured with a straightedge. To serve as the skin section replacement, cut a sheet of 3-ply mahogany poplar core plywood (Spec. AN-NN-P511A) of the same thickness, length, and width as the removed skin *(see Figure 69)*. The replacement skin section must be cut so that the wood grain direction will be laid 45 degrees to the vertical frames. If any appreciable bending in the replacement skin section is necessary in order to match the original skin contour, soak the plywood sheet in hot water, bend over a form, and clamp in place until dry. The form over which the plywood is to be bent should be made so as to allow for some springback after the plywood has dried. Placing the plywood in an atmosphere of saturated steam is the most desirable method for softening the plywood, but the method is a more difficult process to manage. With a spokeshave, scarf the edges of the skin replacement section to match the prepared edges of existing skin. This replacement skin section must fit the scarf edges of the original skin smoothly and snugly without wrinkles, bulges, or gaps. Apply one coat of casein glue (Spec. C-G-456) to both surfaces of the scarf-cut joint and to any interior structure contact surface of the replacement skin. Press the replacement skin into position and make certain the placement is correct. After the replacement skin is pressed into place, avoid any unnecessary adjusting and sliding of the skin section, as this action tends to spread the glue, thus producing a poor joint. To prevent the skin from slipping under subsequent operations, drive three No. 20 gage nails through the center of the skin splice, leaving the nail heads protruding sufficiently to allow subsequent removal of the nails. To provide gluing pressure for the skin scarf joint, secure the scarf joint with nailing strips of 1/4 x 1/2 x 12 soft pine. Nail the strips over the scarf joint with 5/8-inch No. 20 gage nails spaced at 1-1/2 inches on centers *(see Figure 60)*. To prevent bulges, nail first at the center of the scarf joint length, and then progress outward from the center. Use sufficient nailing strips to cover entirely the scarf joint area. For example, if the width of the scarf joint is 7/8-inch, use two sections of 1/2-inch wide nailing strips. After several of the strips are secured, remove the three nails previously driven to secure the skin panel. Allow the scarf joint area to remain under pressure for about 4 hours; then remove the nails and the nailing strips. The replacement skin section must be free of wrinkles, bulges, and joint gaps. Scrape off the excess glue from around the scarf joints. To the inside and outside of the skin panel replacement, apply two coats of sealer (AAF Spec. 14113), allowing a drying time of one hour for each coat. To the outside of the skin panel only, apply one coat of unthinned filler (NAA Formula No. 1398 or equivalent) by rubbing the filler on the wood with a cloth, the direction of the strokes being across the grain. Allow the filler to remain on the outside surface from 2 to 5 minutes, then remove the excess by wiping across the grain with excelsior or with a burlap rag. To the outside of the skin section, brush on one coat of surfacer (AAF Spec. 14115) and allow to dry for 6 hours. Lightly sand the surface with fine sandpaper. To the inside of the skin section replacement, match adjacent surfaces with either one coat of cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98) as required. To the outside of the skin section replacement, apply one coat of aircraft enamel (Spec. 3-98) pigmented with 16 ounces of aluminum paste (Spec. AN-TT-A-461) per gallon of unthinned enamel.

72. BULKHEAD PLYWOOD WEBS.

The bulkhead webs consist of thin 3-ply mahogany stiffened with spruce members *(see Figure 50)*. The webs may be repaired as outlined in the paragraphs pertaining to the repair of the plywood skin.

73. WOODEN REAR FUSELAGE FORMERS.

The wooden rear fuselage formers are formed

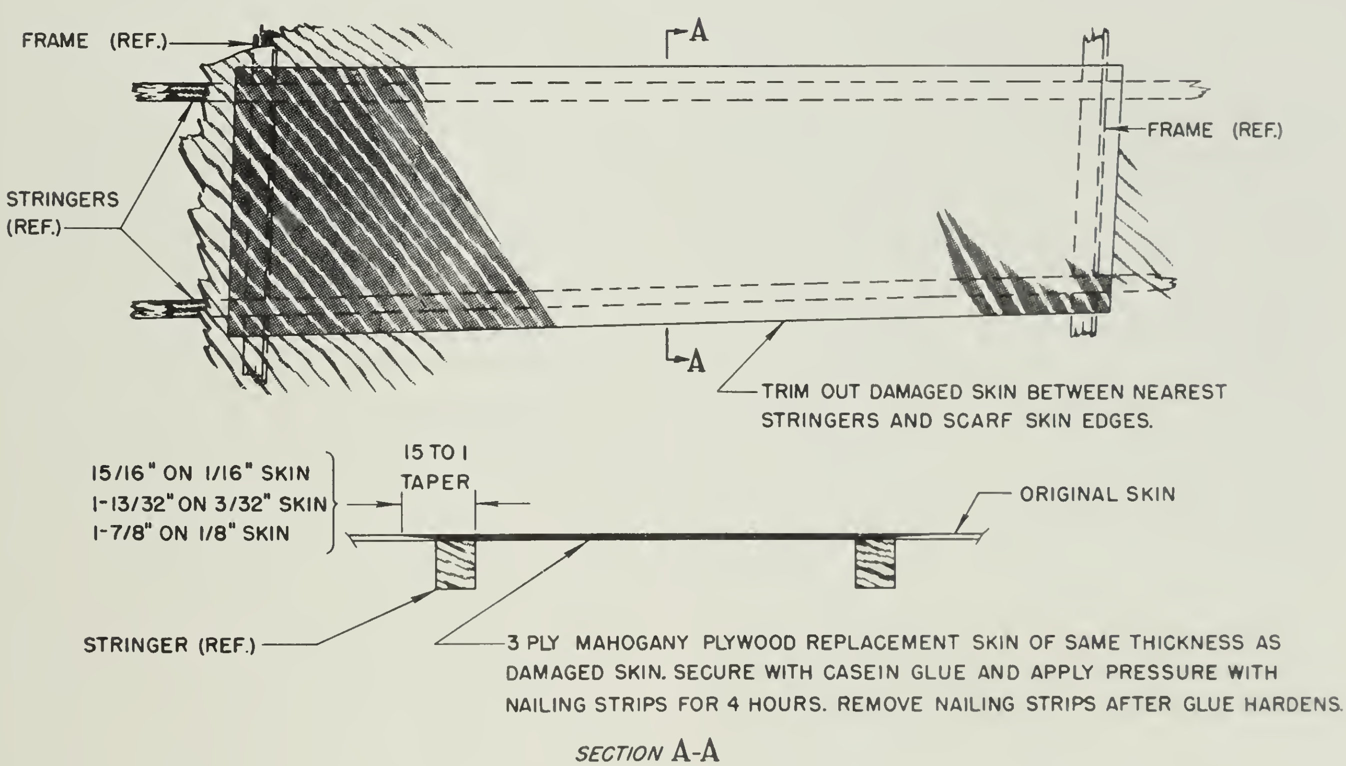


Figure 74—Plywood Skin Section Replacement

of a narrow 3-ply mahogany poplar core web with small spruce tricleats glued along the edges. If the formers become damaged, they may be repaired by splicing in a new former section as shown *(see Figure 75)*. At each side of the damage, cut the web section with a small fine-toothed saw and remove the damaged web. Remove any of the damaged spruce tricleats adjacent to the skin. Scarf-cut the damaged spruce tricleat section located along the inner side of the former, making the cut at a 3-inch taper. Cut and form new spruce (Spec. AN-S-6) tricleat sections to match the spruce tricleats previously removed. Carefully plane down the scarf cuts of the inner tricleat joint so that the cuts are perfectly smooth when measured with a straightedge, and so that they form a perfect fit when assembled. With a spokeshave, scarf-cut the edges of the original web *(see Figure 73)* at a 15-to-1 taper, which amounts to a 15/16-inch taper on this 1/16-inch thick plywood *(see Figure 75)*. Apply one coat of casein glue (Spec. C-G-456) to the skin and the outer replacement tricleats, and secure the tricleats to the skin. To provide gluing pressure for the spruce tricleats, apply nailing strips over the joint area on the outside skin,

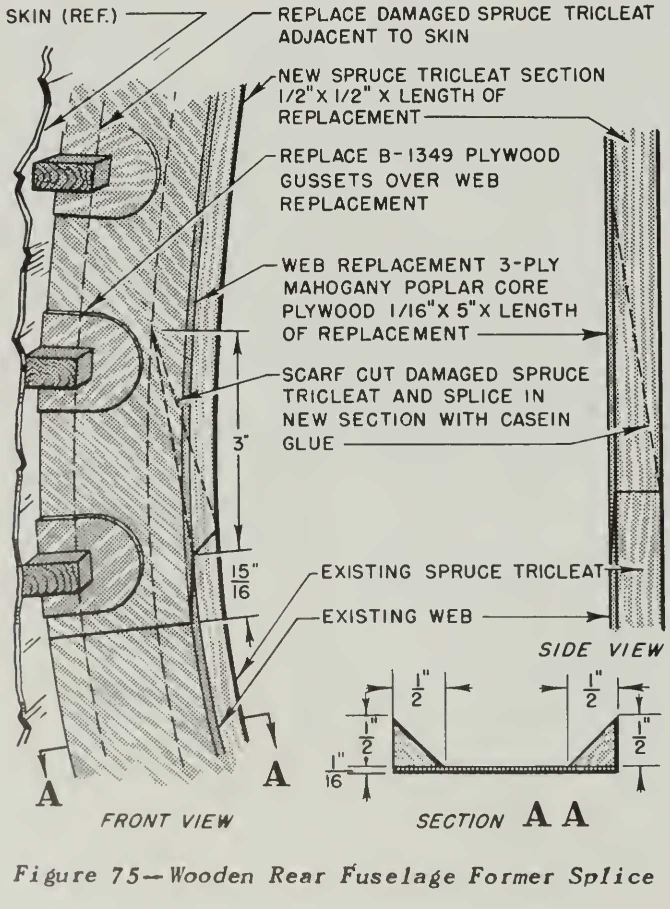


Figure 75—Wooden Rear Fuselage Former Splice

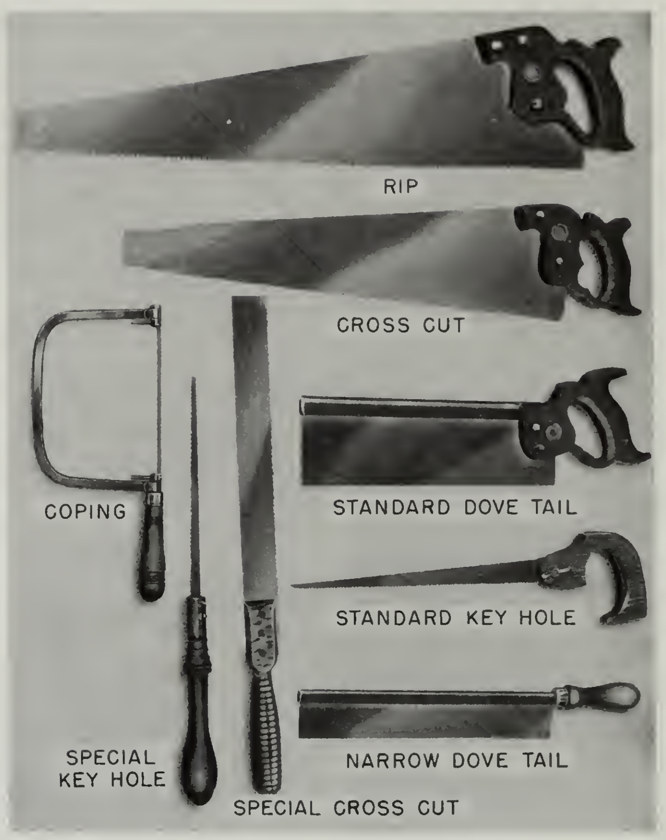


Figure 76—Types of Saws

back up the tricleats, and then nail the skin to the tricleats. Apply one coat of casein glue (Spec. C-G-456) to both surfaces of the inner spruce tricleat scarf joint, assemble the joint, and clamp in place. Allow the glue to dry for at least 4 hours, and then remove the clamps and the nailing strips. Apply one coat of casein glue to the replacement web section and to scarf joint contact surfaces of the tricleats and replacement web. Press the replacement web into the proper position on the former, and clamp the glued surfaces. Allow the glue to dry for at least 4 hours; then remove the clamps. To the entire added material, apply one coat of sealer (AAF Spec. 14113) and allow to dry for one hour. To all the added material except the mahogany web and the end-grain surfaces of the outer tricleats, apply another coat of sealer (AAF Spec. 14113). To the end-grain surfaces of the outer tricleats and mahogany web, apply one coat of unthinned filler (NAA Formula No. 1398 or equivalent) by rubbing on with a cloth. Allow the filler to remain on the surface from 2 to 5 minutes, and then remove the excess by wiping with excelsior or with a burlap rag. Match adjacent finish with either one coat of cellulose nitrate

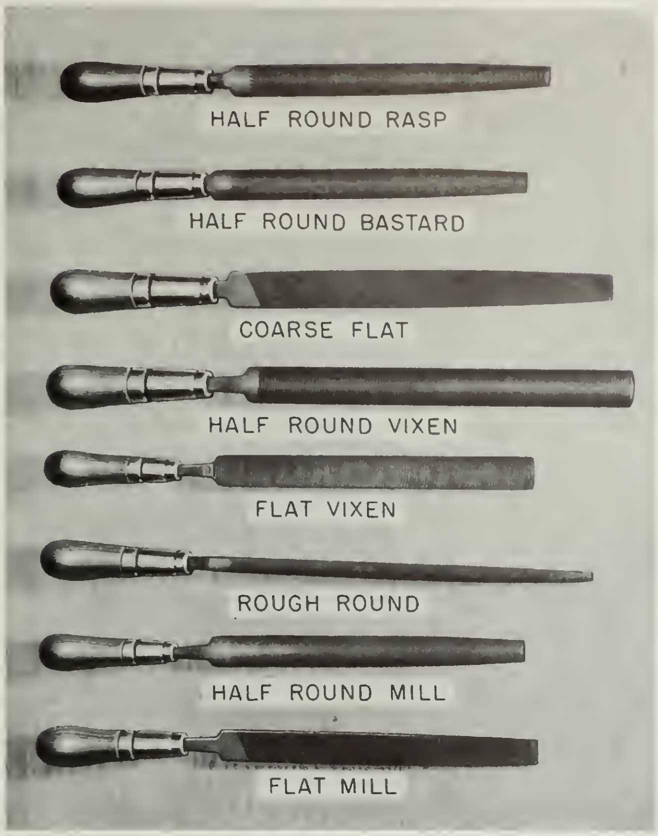


Figure 77—Types of Files for Wood

lacquer (Spec. AN-TT-L-51) or one coat of aircraft enamel (Spec. 3-98) as required.

74. GENERAL FINISH REQUIREMENTS FOR WOOD.

Prior to gluing or finishing any wood surface, thoroughly clean the surface to remove sawdust, grease, and any other foreign matter, from the wood. Wherever possible, accomplish all gluing prior to finishing, and remove all excess glue from visible interior or exterior parts before applying any finish. Whenever the peculiarity of the part is such that it is more conveniently glued subsequent to finishing, mask all areas to be glued with Scotch Tape (or suitable equivalent). No adhesive must be left on the wood when the tape is removed. Apply doped sealing tape to all areas previously taped. Before attaching metal fittings, supports, etc., protect them from direct contact with the wood by applying two coats of sealer, or one coat of sealer and one coat of filler, to the wood under the fitting. For convenience, the wood surfaces may be divided into three groups, in order to facilitate the specification of finishes. The wood surface groups and the corresponding finish for each are as follows *(see Figure 79)*.

75. WOOD FINISH MATERIALS.

The following materials are required in the application of a protective finish to the wood repair material:

MATERIAL	SPECIFICATION
ENAMEL, CAMOUFLAGE	14109, AAF
SEALER, WOOD, LIQUID	14113, AAF
SURFACER, WOOD, LIQUID	14115, AAF
ENAMEL, AIRCRAFT	3-98, U.S. ARMY
PASTE, ALUMINUM	AN-TT-A-461, ARMY-NAVY
DOPE, CELLULOSE NITRATE, CLEAR	AN-TT-D-514, ARMY-NAVY
LACQUER, CELLULOSE NITRATE	AN-TT-L-51, ARMY-NAVY
FILLER, WOOD	NAA FORMULA NO. 1398 OR EQUIVALENT

76. WOOD REPAIR MATERIALS.

The following is a summation of the types of wood and glue that may be required in the repair of the wooden rear fuselage section. The quantities of wood required for a particular

Figure 78—Types of Planes for Wood

WOOD SURFACE DESIGNATION	WHERE LOCATED	REQUIRED FINISH
Type I	This group includes the interior of closed members or surfaces normally remaining unseen, such as stabilizer interiors, hollow longerons, the underside of floorboards, etc.	Apply two brush coats of sealer (AAF Spec. 14113), allowing each coat to dry one hour. The second coat of sealer may be aluminized with 8 ounces of aluminum paste per gallon of unthinned sealer. Do not thin the sealer to less than 25% nonvolatile content. To assure complete coverage, and as a guide in subsequent sanding, it is permissible to add a suitable dye to the sealer.
Type II	This group includes the interior open surfaces, parts, and members visible to occupants of the aircraft, such as the fuselage compartments and attaching parts.	As outlined for Type I surfaces, apply one coat of sealer (AAF Spec. 14113) and allow to dry for one hour. Apply a second coat of unaluminized sealer to all surfaces except the exposed open end grains and mahogany. To mahogany and to exposed open end grains, apply one coat of filler (NAA Formula No. 1398 or equivalent) by rubbing with a cloth. The direction of the strokes shall be across the grain. Allow the filler to remain on the surface from 2 to 5 minutes, and then remove the excess with excelsior or with a burlap rag. Tape all end-grain surfaces with regulation glider fabric or equivalent, predoping the tape with one heavy coat of clear dope (Spec. AN-TT-D-514) and allowing to dry. Tape shall be pinked on all edges and shall give a minimum lap of 1/2-inch onto the straight-grain surface. Apply a second coat of dope to the tape and immediately press the tape down firmly on exposed end grain, eliminating all air pockets. Apply an additional coat of clear dope if the nap of the tape is not satisfactorily filled and sealed down. Doped tapes will not adhere satisfactorily to finished areas. End-grain surfaces not suitable for taping should have an extra coat of sealer in addition to the regular finish. When finishing at drain holes and other inaccessible places, use a stiff wire covered with cloth or use an ordinary pipe cleaner. Allow tape to dry before applying any finish. Match adjacent surfaces with either one coat of cellulose nitrate lacquer (Spec. AN-TT-L-51) or aircraft enamel (Spec. 3-98). The finish coat shall be of sufficient thickness to give complete hiding of the finishing materials.
Type III	This group includes the exterior surfaces, parts, and members, such as surfaces exposed to the weather and to view from the outside.	Apply two coats of unaluminized sealer (AAF Spec. 14113) as outlined for Type I surfaces. Apply one coat of filler (NAA Formula No. 1398 or equivalent) as outlined for Type II surfaces. Apply one coat of surfacer (AAF Spec. 14115) and allow to air-dry for 6 hours. With dry sandpaper, sand the surfacer after it has been thoroughly dried. Sand so as to produce the minimum possible thickness of the surfacer coat but do not sand to the filler coat beneath. Match adjacent finish with one coat of clear enamel (Spec. 3-98) pigmented with 16 ounces of aluminum paste (Spec. AN-TT-A-461) per gallon of unthinned enamel.

Figure 79—Finish Requirements for Wood

repair must be determined locally with reference to the applicable repair procedure outlined in this Section.

MATERIAL	SPECIFICATION
PLYWOOD, 3-PLY MAHOGANY, POPLAR CORE	AN-NN-P-511A
PLYWOOD, 3-PLY DOUGLAS FIR, POPLAR CORE	AN-NN-P-511A
PLYWOOD, 3-PLY DOUGLAS FIR, DOUGLAS FIR CORE	AN-NN-P-511A
PLYWOOD, 3-PLY RED GUM, RED GUM OR POPLAR CORE	AN-NN-P-511A
SPRUCE	AN-S-6
POPLAR, YELLOW	AN-P-17
HEMLOCK, WESTERN	AN-H-4
FIR, NOBLE	AN-F-6
COMPREGWOOD	IMPREGNATED AND COMPRESSED LAMINATED MAPLE OF DENSITY 80 TO 85 LBS. PER CU. FT. MAY BE PURCHASED FROM FORMICA INSULATING CO., CINCINNATI, OHIO, OR EQUIVALENT.
GLUE, CASEIN	C-G-456 (COMMERCIALLY OBTAINABLE AS "MONITE GLUE", MANUFACTURED BY MONITE WATERPROOF GLUE CO., 1626 N. SECOND ST., MINNEAPOLIS, MINN. ANOTHER COMMERCIAL GLUE CONFORMING TO THE MATERIAL SPECIFICATION IS "LAUXEIN WATERPROOF GLUE #888", MANUFACTURED BY I.F. LAUCKS, INC., SEATTLE, WASHINGTON.)

77. WOOD REPAIR TOOLS.

To repair wooden structures adequately, all or part of the following wood-working tools may be required:

TOOL	REMARKS
CLAMPS	JORGENSEN, "C", BAR, HANDSCREW
HAMMERS	SMALL BALL PEEN OR CLAW AS REQUIRED
SAWS	MITER, BOX, RIP, KEYHOLE, AND CROSSCUT, AS REQUIRED
BITS	AS REQUIRED FOR DRILLING VENTILATING HOLES
CHISELS	WIDTH AND SHAPE AS REQUIRED
PLANES	TYPE, LENGTH, AND WIDTH AS REQUIRED
SQUARES	TYPE AND SIZE AS REQUIRED
MEASURING TOOLS	RULES, SCALES, DIVIDERS, COMPASSES, AND MARKING GAGES
FILES	WOOD FILES, RASPS, ETC., AS REQUIRED
DRAWKNIFE	
SPOKESHAVE	

SECTION 3
FIXED SURFACES

1. WING CONSTRUCTION.

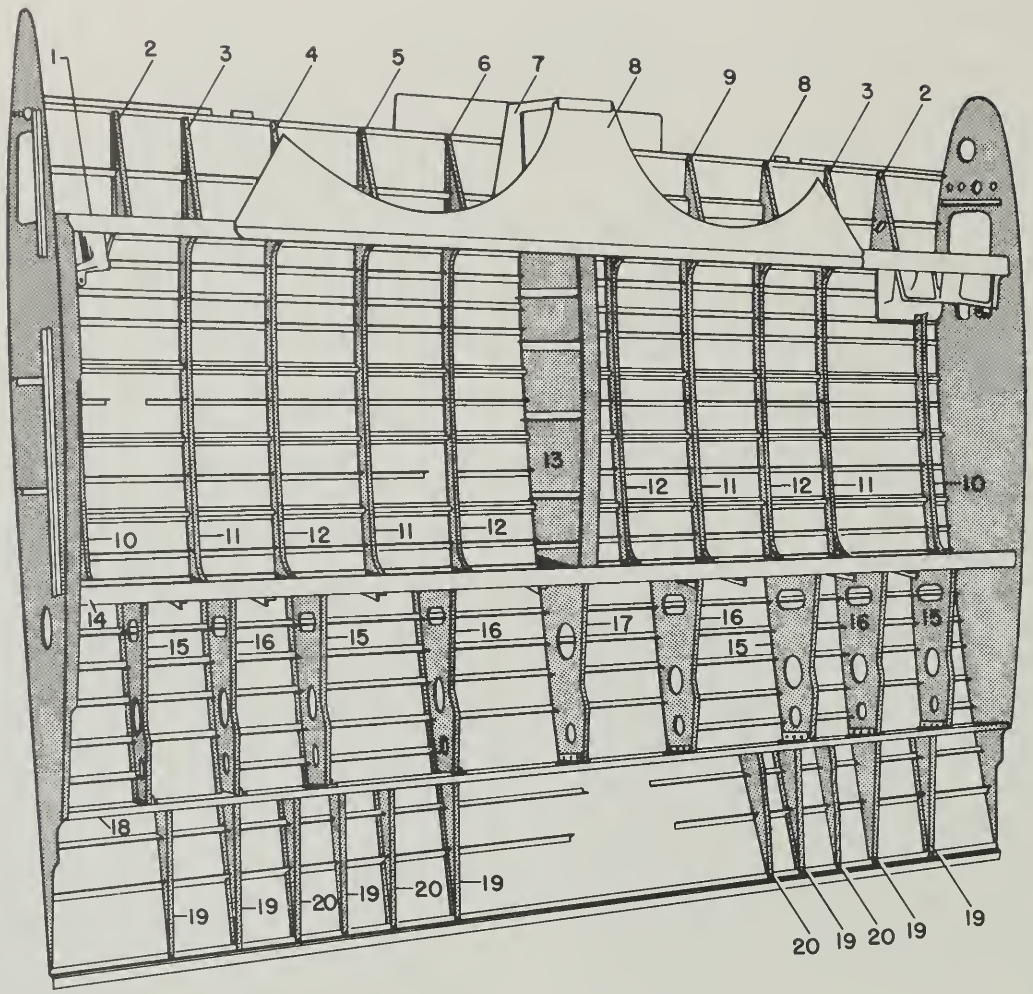
The low wing is of full-cantilever, stressed skin design and consists of a detachable centersection and detachable outer wing sections. The entire wing has a span of 42 feet 1/4-inch and an area of 253.72 square feet. The structure is formed from 24ST alclad sheet and 24ST aluminum alloy extrusions and sheet, consisting of spars, power pressed ribs, and skin stiffened in a spanwise direction by stringers. The outer wing sections and the centersection are joined by external bolting angles along the upper and lower skin surfaces. A plate rib at each joint distributes shear from the single outer panel main spar to the two centersection spars. The centersection is of constant chord design and has an incidence of plus 2 degrees, and no dihedral angle. Each outer wing incorporates an angle of incidence of plus 2 degrees at the root which gradually changes to an angle of incidence of 0 degrees at the tip. Each outer wing is bolted to the centersection at a dihedral angle of 5 degrees 41 minutes. The skin panels vary from .020 to .064 inch in thickness and are lap jointed. Type AD rivets (A17S aluminum) are used for assembly, all skin riveting being flush to approximately 33% chord on the top skin and to 10% chord on the bottom skin. The notation "wing station" indicates the distance in inches measured from the centerline of the airplane to any position outboard of the centerline. The term "station" as used in reference to each outer wing may signify the distance in inches outboard of the wing joint, or outboard of the centerline of the airplane as noted *(see Figure 9)*. The centersection is attached to the underside of the welded steel fuselage tubing truss by bolt fittings at the two spars of the centersection. For quick reference, the part numbers of the rib and spar structure of the centersection are illustrated *(see Figure 1)*. The part numbers of the rib and spar structure of the outer wing are also shown *(see Figure 2)*. The part numbers of the centersection fuel tank compartment doors are shown *(see Figure 3)*.

2. HORIZONTAL STABILIZER CONSTRUCTION.

The horizontal stabilizer is a full-cantilever aluminum alloy structure having a symmetrical airfoil. The stabilizer has a total area of 28.34 square feet, including 5.33 square feet of fuselage area. The stabilizer consists of two interchangeable sections bolted together at the center. Each section of the stabilizer consists essentially of two spars, regularly spaced ribs, and external skin stiffened by spanwise stringers *(see Figure 4)*. The rear spar of each of the stabilizer sections carries all shear loads as a cantilever beam. The torque about the rear spar is carried as shear in the skin to the inboard rib where the shear is reacted by a vertical couple at the spar attaching points. On the later AT-6C Series Airplanes, the aluminum horizontal stabilizer is substituted by a stabilizer constructed entirely of wood *(see Figure 5)*. The basic construction of the wooden horizontal stabilizer differs little from that of the aluminum structure. The only noticeable difference is the four additional nose ribs which are attached to the front spar to give additional support to the plywood leading edge. The spar caps are made from compregwood with the webs of mahogany-poplar plywood *(see Figure 6)*. The rib webs are also of mahogany-poplar plywood with spruce tricleat caps. The stringers are of spruce and are held securely in position in the rib cutouts by means of small wedges and glue. Ventilating holes are provided in the web of all ribs and these holes are reinforced by a plywood washer which is glued around the hole *(see Figure 5)*. The 3-ply, mahogany-poplar plywood skin is glued to the stringers, ribs, and spar caps by means of casein glue. The two sections of the wood stabilizer are bolted at the rear spar through aluminum alloy fittings *(see Figure 6)*. The attaching fitting on the front spar consists of a light aluminum sheet bolted to the spar *(see Figure 7)*.

3. VERTICAL STABILIZER CONSTRUCTION.

The vertical stabilizer is a full-cantilever aluminum alloy structure having an area of 5.33 square feet and a normal setting of 1-3/4 degrees to the left. The stabilizer is constructed of 24ST alclad sheet consisting of two spars, several ribs, and skin stiffened in a spanwise direction with stringers *(see Figure 8)*. The rear spar is the main load-bearing member of the structure and is formed of a flat 24ST sheet, with U flanges formed along the length. The leading edge portion of the



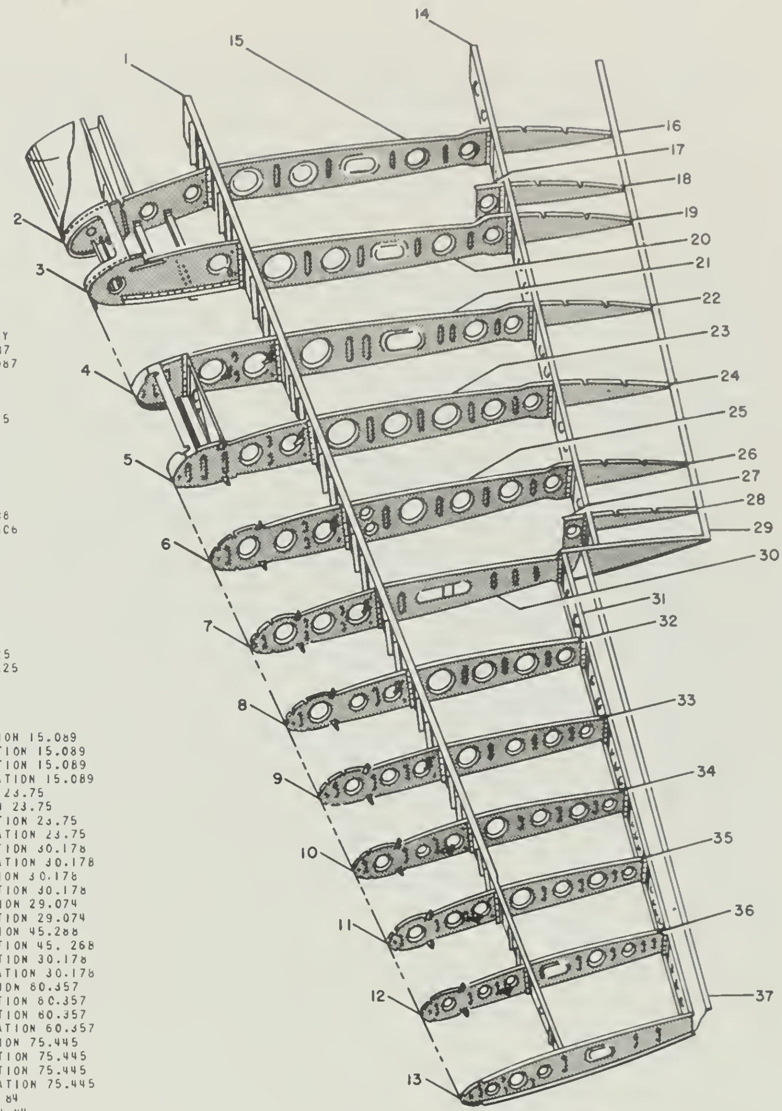
NORTH AMERICAN AVIATION PART NUMBERS

	CONTRACT NOS. AT-6A 77	AT-68 84	AT-6C 88	DRAWING NOS.	TITLE
1.	77	77	8B	13003	FRONT SPAR
2.	66	66	66	13048	LEFT RIB - STATION 46-1/8
	66	66	68	13048-1	RIOHT RIB - STATION 46-1/8
3.	66	66	66	13027	LEFT RIB - STATION 39
	66	66	66	13027-1	RIGHT RIB - STATION 39
4.	77	77		13030-3	RIGHT RIB - STATIONS 30 AND 10
			66	13026-1	RIGHT RIB - STATION 30
			77	13030-5	RIGHT RIB - STATION 10
5.	77	77		13030	RIGHT RIB - STATION 20
			66	13026-5	RIGHT RIB - STATION 20
6.	77	77	77	13036	RIGHT CENTER RIB
7.	66	66	66	1302B-4	LEFT CENTER RIB
8.	66	66	66	13026-6	LEFT RIB - STATIONS 30 AND 10
9.	66	66	66	13026	LEFT RIB - STATION 20
10.	66	84	B4	13041	LEFT RIB - STATION 53-1/2
	66		84	13042-5	RIGHT RIB - STATION 53-1/2
		84		13041-5	RIGHT RIB - STATION 53-1/2
11.	66	84	84	13042	LEFT RIB - STATIONS 39 AND 20
	66			13042-5	RIGHT RIB - STATIONS 39 AND 20
		84	84	13042-1	RIGHT RIB - STATIONS 39 AND 20
12.	66			13040	LEFT RIB - STATIONS 30 AND 10
	66			13040-5	RIGHT RIB - STATIONS 30 AND 10
		84	84	13042-2	LEFT RIB - STATIONS 30 AND 10
		84	84	13042-3	RIGHT RIB - STATIONS 30 AND 10
13.	66	84	84	13029	CENTER RIB - STATION 0
14.	66	66	66	13004	REAR SPAR
15.	66	66	66	13190	LEFT RIB - STATIONS 48 AND 30
	66	66	66	13190-1	RIGHT RIB - STATIONS 4B AND 30
16.	66	66	66	13192	LEFT RIB - STATIONS 39 AND 15
	66	66	66	13192-1	RIGHT RIB - STATIONS 39 ANO 15
17.	66	66	66	13193	CENTER RIB - STATION 0
18.	66	66	66	13205	WING FLAP SPAR
19.	55	55		13194	LEFT RIB - STATIONS 46-15/16, 39, ANO 28-15/16
	55	55		13194-1	RIGHT RIB - STATIONS 46-15/16, 39, 28-15/16 AND 15
			55	13194-5	RIGHT RIB - STATIONS 46-15/16 AND 28-15/16
			55	13194	LEFT RIB - STATIONS 46-15/16 ANO 28-15/16
			55	13194-1	RIGHT RIB - STATIONS 39 AND 15
			55	13194	LEFT RIB - STATIONS 46-15/16 AND 2B-15/16
20.	55	55	55	13294	LEFT RIB - STATIONS 33-21/32 AND 23-15/16
	55	55	55	13294-1	RIGHT RIB - STATIONS 33-21/32 AND 23-15/16

Figure 1—Centersection Rib and Spar Part Numbers

NORTH AMERICAN AVIATION PART NUMBERS

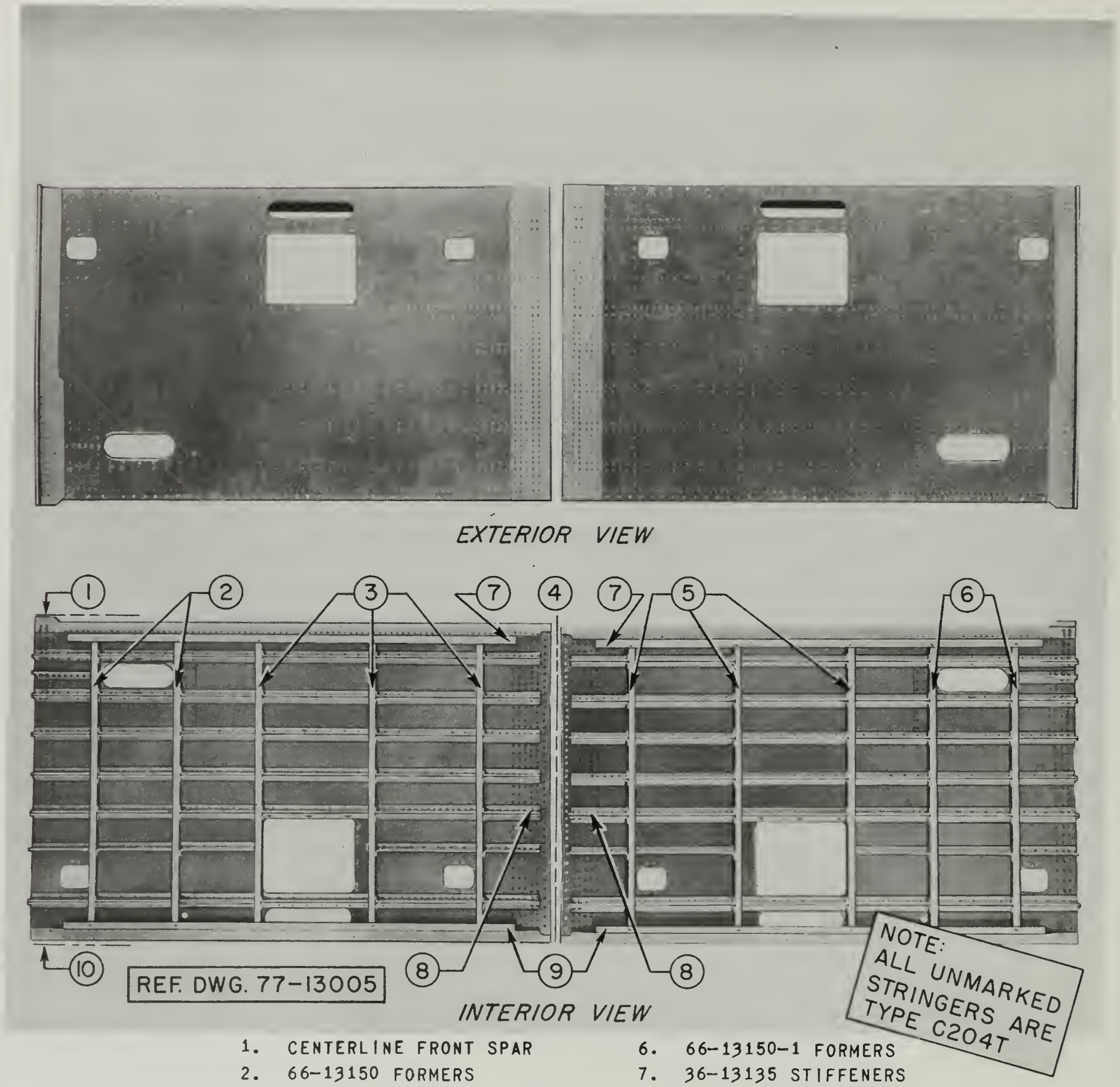
	CONTRACT NOS. AT-6A 77	AT-6B 84	AT-6C 88	DRAWING NOS.	TITLE
1.	55	64	84	14C04	LEFT OUTER WING SPAR ASSEMBLY
	55	84	88	14C04-1	RIGHT OUTER WING SPAR ASSEMBLY
2.	54	54	88	14C15	LEFT NOSE RIB - STATION 15.087
	54	54	88	14C15-1	RIGHT NOSE RIB - STATION 15.087
3.	55			14218	LEFT NOSE RIB - STATION 30
	55			14218-1	RIGHT NOSE RIB - STATION 30
		84	84	14218	LEFT NOSE RIB - STATION 27.25
		84	84	14C12	RIGHT NOSE RIB - STATION 27.25
4.	52	52	52	14217	LEFT NOSE RIB - STATION 45
	52	52	52	14217-1	RIGHT NOSE RIB - STATION 45
5.	52	52	52	14218	LEFT NOSE RIB - STATION 60
	52	52	52	14218-1	RIGHT NOSE RIB - STATION 60
6.	52	52	52	14219	LEFT NOSE RIB - STATION 75
	52	52	52	14219-1	RIGHT NOSE RIB - STATION 75
7.	52	52	52	14220	LEFT NOSE RIB - STATION 89.808
	52	52	52	14220-1	RIGHT NOSE RIB - STATION 89.606
8.	52	52	52	14221	LEFT NOSE RIB - STATION 105
	52	52	52	14221-1	RIGHT NOSE RIB - STATION 105
9.	52	52	52	14222	LEFT NOSE RIB - STATION 120
	52	52	52	14222-1	RIGHT NOSE RIB - STATION 120
10.	52	52	52	14223	LEFT NOSE RIB - STATION 135
	52	52	52	14223-1	RIGHT NOSE RIB - STATION 135
11.	52	52	52	14224	LEFT NOSE RIB - STATION 150
	52	52	52	14224-1	RIGHT NOSE RIB - STATION 150
12.	52	52	52	14225	LEFT NOSE RIB - STATION 167.25
	52	52	52	14225-1	RIGHT NOSE RIB - STATION 167.25
13.	52	52	52	14226-2	LEFT RIB - STATION 186.351
	52	52	52	14228-5	RIGHT RIB - STATION 186.351
14.	55	84	64	14045	LEFT OUTER WING FLAP SPAR
	55	84	84	14045-1	RIGHT OUTER WING FLAP SPAR
15.	55	84	84	14245	LEFT INTERMEDIATE RIB - STATION 15.089
	55	84	84	14245-1	RIGHT INTERMEDIATE RIB - STATION 15.089
16.	55	55	55	14231	LEFT TRAILING EDGE RIB - STATION 15.089
	55	55	55	14231-1	RIGHT TRAILING EDGE RIB - STATION 15.089
17.	55	55	55	14252	LEFT REINFORCEMENT - STATION 23.75
	55	55	55	14252-1	RIGHT REINFORCEMENT - STATION 23.75
18.	55	55	55	14250	LEFT TRAILING EDGE RIB - STATION 23.75
	55	55	55	14250-1	RIGHT TRAILING EDGE RIB - STATION 23.75
19.	55	84	84	14232	LEFT TRAILING EDGE RIB - STATION 30.178
	55	84	84	14232-1	RIGHT TRAILING EDGE RIB - STATION 30.178
20.	55			14248	LEFT INTERMEDIATE RIB - STATION 30.178
	55			14246-1	RIGHT INTERMEDIATE RIB - STATION 30.178
		84	84	14248	LEFT INTERMEDIATE RIB - STATION 29.074
		84	84	14248-1	RIGHT INTERMEDIATE RIB - STATION 29.074
21.	55	84	84	14247	LEFT INTERMEDIATE RIB - STATION 45.288
	55	84	84	14247-1	RIGHT INTERMEDIATE RIB - STATION 45. 268
22.	55	55	55	14233	LEFT TRAILING EDGE RIB - STATION 30.178
	55	55	55	14233-1	RIGHT TRAILING EDGE RIB - STATION 30.178
23.	55	84	84	14248	LEFT INTERMEDIATE RIB - STATION 60.357
	55	84	84	14248-1	RIGHT INTERMEDIATE RIB - STATION 60.357
24.	55	55	55	14234	LEFT TRAILING EDGE RIB - STATION 60.357
	55	55	55	14234-1	RIGHT TRAILING EDGE RIB - STATION 60.357
25.	55	55	55	14249	LEFT INTERMEDIATE RIB - STATION 75.445
	55	55	55	14249-1	RIGHT INTERMEDIATE RIB - STATION 75.445
26.	55	55	55	14235	LEFT TRAILING EDGE RIB - STATION 75.445
	55	55	55	14235-1	RIGHT TRAILING EDGE RIB - STATION 75.445
27.	55	55	55	14253	LEFT REINFORCEMENT - STATION 84
	55	55	55	14253-1	RIGHT REINFORCEMENT - STATION 84
28.	55	55	55	14251	LEFT TRAILING EDGE RIB - STATION 84
	55	55	55	14251-1	RIGHT TRAILING EDGE RIB - STATION 84
29.	55	55	55	14104	LEFT TRAILING EDGE STRIP
	55	55	55	14104-1	RIGHT TRAILING EDGE STRIP
30.	55	55	55	14238	LEFT TRAILING EDGE RIB - STATION 90.139
	55	55	55	14238-1	RIGHT TRAILING EDGE RIB - STATION 90.139
31.	54	54	54	14051	LEFT WING AILERON CUTOUT CHANNEL SPAR
	54	54	54	14051-1	RIGHT WING AILERON CUTOUT CHANNEL SPAR
32.	52	52	52	14237	LEFT TRAILING EDGE RIB - STATION 105
	52	52	52	14237-1	RIGHT TRAILING EDGE RIB - STATION 105
33.	52	52	52	14238	LEFT TRAILING EDGE RIB - STATION 120
	52	52	52	14238-1	RIGHT TRAILING EDGE RIB - STATION 120
34.	55	55	55	14239	LEFT TRAILING EDGE RIB - STATION 135.802
	55	55	55	14239-1	RIGHT TRAILING EDGE RIB - STATION 135.802
35.	52	52	52	14240	LEFT TRAILING EDGE RIB - STATION 150
	52	52	52	14240-1	RIGHT TRAILING EDGE RIB - STATION 150
36.	52	52	52	14241	LEFT TRAILING EDGE RIB - STATION 187.25
	52	52	52	14241-1	RIGHT TRAILING EDGE RIB - STATION 187.25
37.	54	54	54	14084	LEFT AILERON SLOT FAIRING
	54	54	54	14084-1	RIGHT AILERON SLOT FAIRING



REF. DWGS.
55-14001-50(-51 RH) AT-6A
84-14001(-1RH) AT-6B
88-14001(-1RH) AT-6C

NOTE:
STRINGERS ARE
NOT SHOWN

Figure 2—Outer Wing Rib and Spar Part Numbers



1. CENTERLINE FRONT SPAR
2. 66-13150 FORMERS
3. 66-13151 FORMERS
4. CENTERLINE AIRPLANE
5. 66-13151-1 FORMERS
6. 66-13150-1 FORMERS
7. 36-13135 STIFFENERS
8. C250T STRINGERS
9. 36-13135-1 STIFFENERS
10. CENTERLINE REAR SPAR

Figure 3--Centersection Fuel Tank Compartment Doors

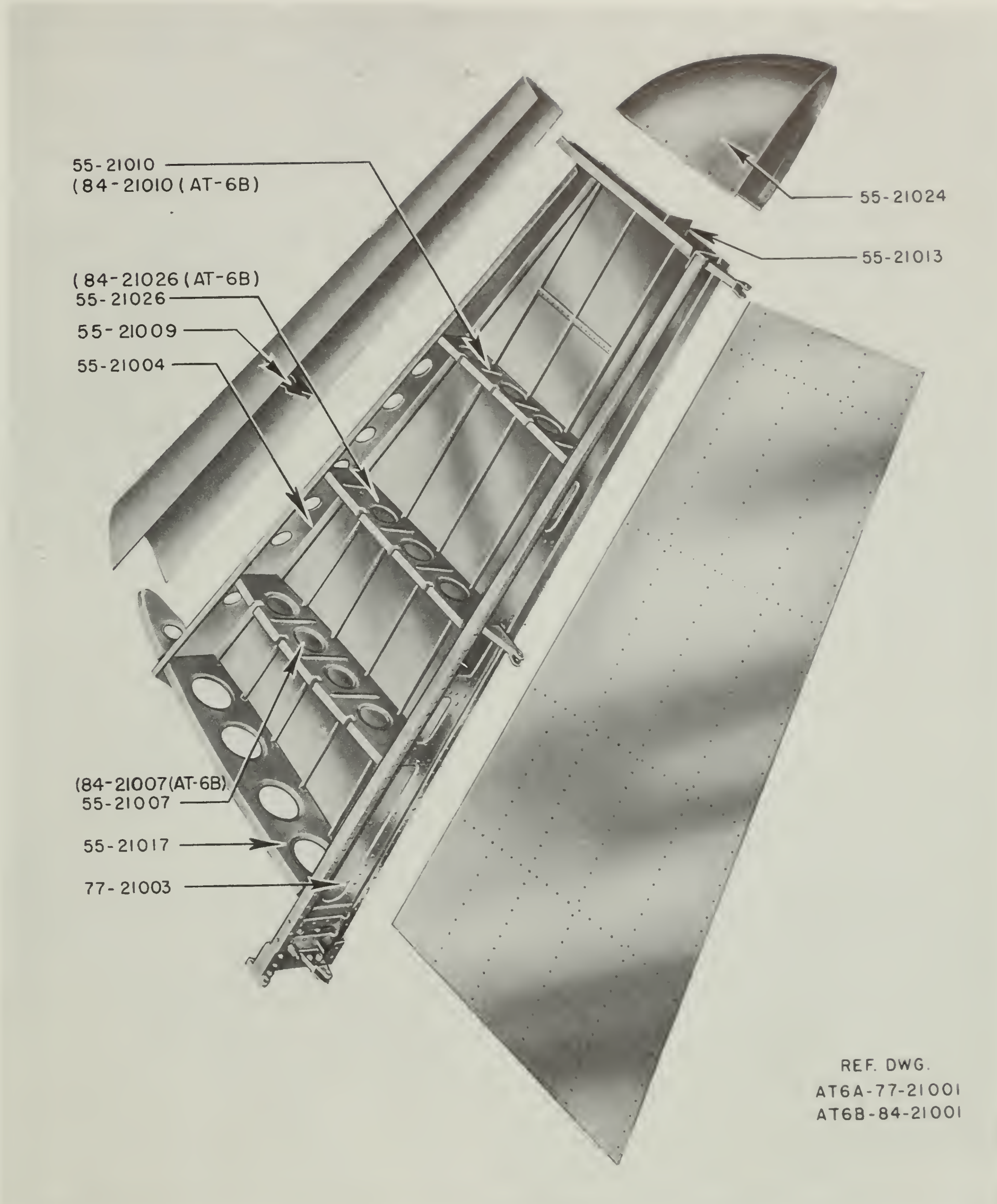
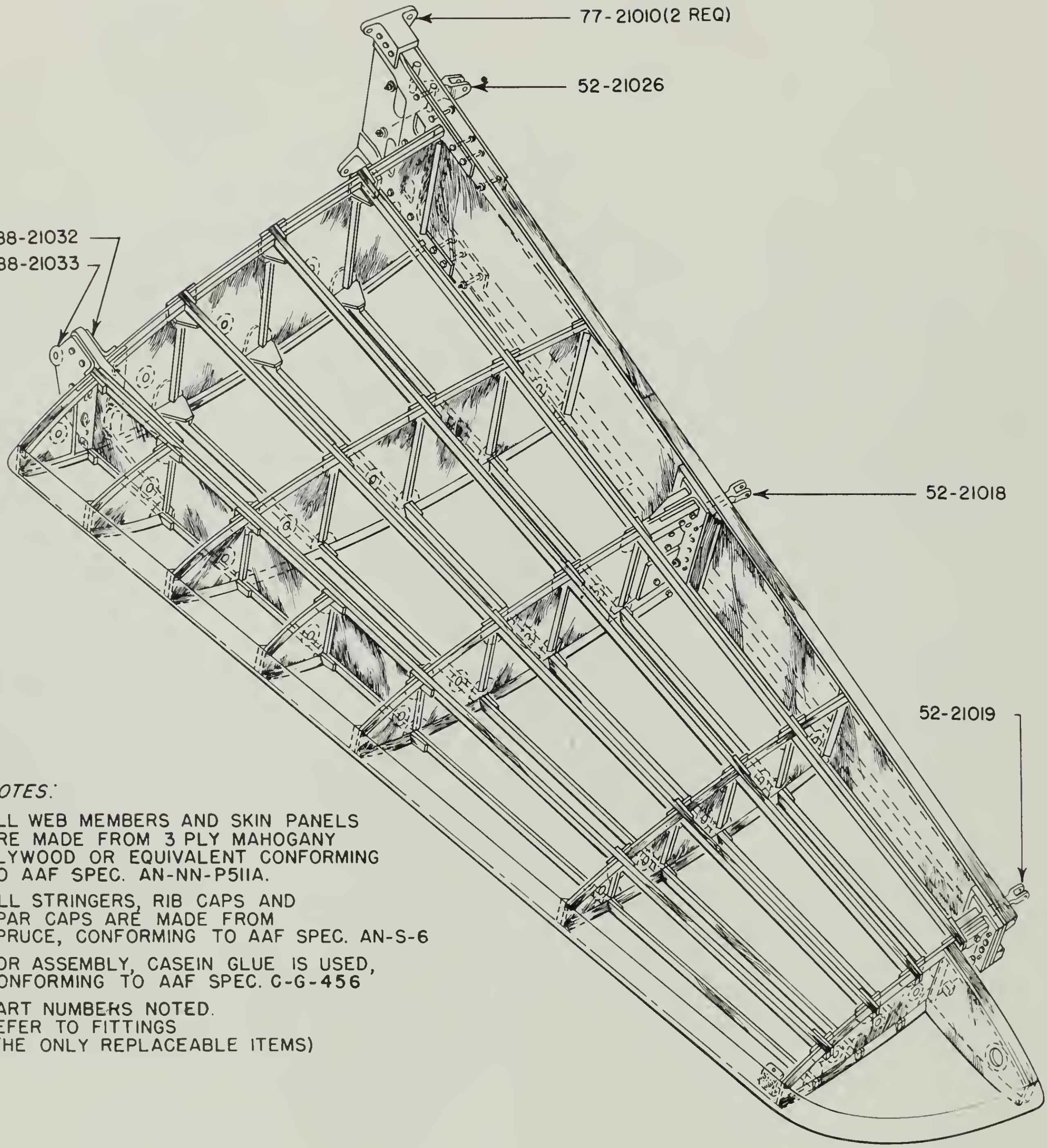


Figure 4—Aluminum Horizontal Stabilizer Structure



NOTES:

1. ALL WEB MEMBERS AND SKIN PANELS ARE MADE FROM 3 PLY MAHOGANY PLYWOOD OR EQUIVALENT CONFORMING TO AAF SPEC. AN-NN-P511A.
2. ALL STRINGERS, RIB CAPS AND SPAR CAPS ARE MADE FROM SPRUCE, CONFORMING TO AAF SPEC. AN-S-6
3. FOR ASSEMBLY, CASEIN GLUE IS USED, CONFORMING TO AAF SPEC. C-G-456
4. PART NUMBERS NOTED. REFER TO FITTINGS (THE ONLY REPLACEABLE ITEMS)

REF. DWG. 88-21001 (AT-6C)

Figure 5—Wooden Horizontal Stabilizer Structure

stabilizer is flush riveted, the remaining portion being riveted with brazier head rivets.

4. STRINGERS - GENERAL.

Most of the stringers used in the construction of the fixed surfaces are extruded shapes of 24ST aluminum alloy material. However, a few rolled shapes of 24ST alclad sheet material are also employed. In the AT-6A wing, the stringer types and locations in the right wing are symmetrical with those in the left wing *(see Figure 9)*. In the AT-6B and AT-6C wing,

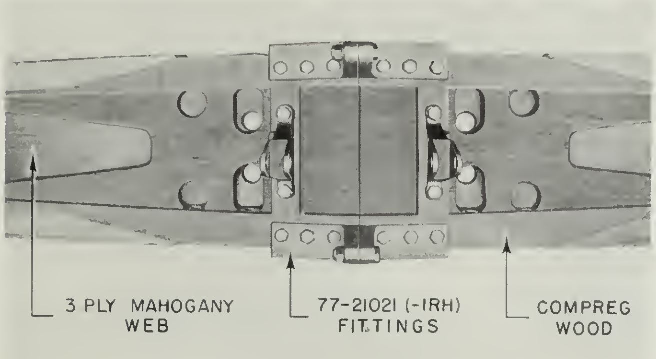


Figure 6—Rear Spar Joint of Wooden Horizontal Stabilizer

the stringer types and locations in the left wing *(see Figure 10)* are symmetrical with those in the right wing except for a small portion of the upper and lower surfaces of the outer wing leading edge at the wing joint *(see Figures 11 and 12)*. On the AT-6A stabilizers, C107LT rolled alclad stringers are employed *(see Figure 13)*. On the AT-6B and AT-6C stabilizers, Type C373LT rolled alclad stringers are employed *(see Figure 13)*. From these reference drawings, determine the type number of the damaged stringer and then refer to the following applicable paragraph for the splice procedure. Those stringer types having a C prefix are North American Aviation, Inc., standard parts. Those stringer types having a K prefix are Aluminum Company of America (Alcoa) die number stringers. For quick reference, the following chart sums up the types of stringers and the approximate quantities involved in the original construction of the fixed surfaces *(see Figure 15)*. For the Alcoa equivalents of North American Aviation, Inc., standard extrusions, see the complete summation of extrusions in Section I.

5. STRINGER REPAIRS - GENERAL.

In repairing damage to stringers, do not make any splices in the bay at the end of the stringer; replace a portion of the damaged stringer and locate the splice outboard of the first bay. If skin rivets through the stringer are spaced at 1-1/4 inches on centers or greater, double the number of existing skin rivets through the stringer at the splice. Skin rivets and splice rivets at the end of the splice must have at least a 3/8-inch edge distance, and all splice members must be in the fully heat-treated condition. Where ribs interfere with the installation of a stringer splice, modify the rib cutout as required. It is to be noted that where damaged stringers are short enough to permit entire removal, the stringers should be replaced. At the root of the outer wing, a C123LT stringer is reinforced with a short length of Type C203T stringer in one location and a C140T in another position. Repairs to these stringer combinations should be made by replacing the short reinforcing stringer and by splicing the C123LT stringer outboard of the doubled location. If any stringers are replaced near the wing joint, file the stringers so that they bear directly upon a straight edge supported by the upper and lower attaching angles. During the reassembly of the wing, where some of the stringers near the wing joint were replaced, it may be necessary to taper the end of the new stringers a very small amount if interference occurs in the wing installation.

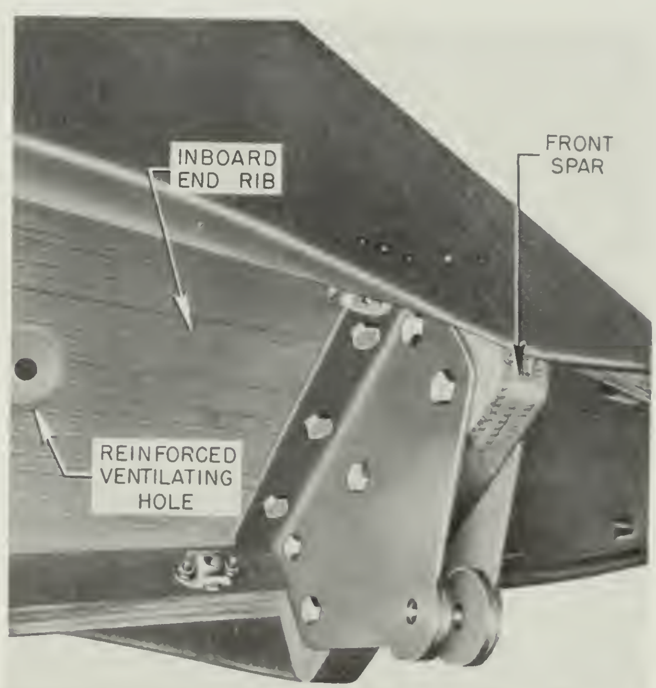


Figure 7—Wooden Horizontal Stabilizer Front Spar Fitting

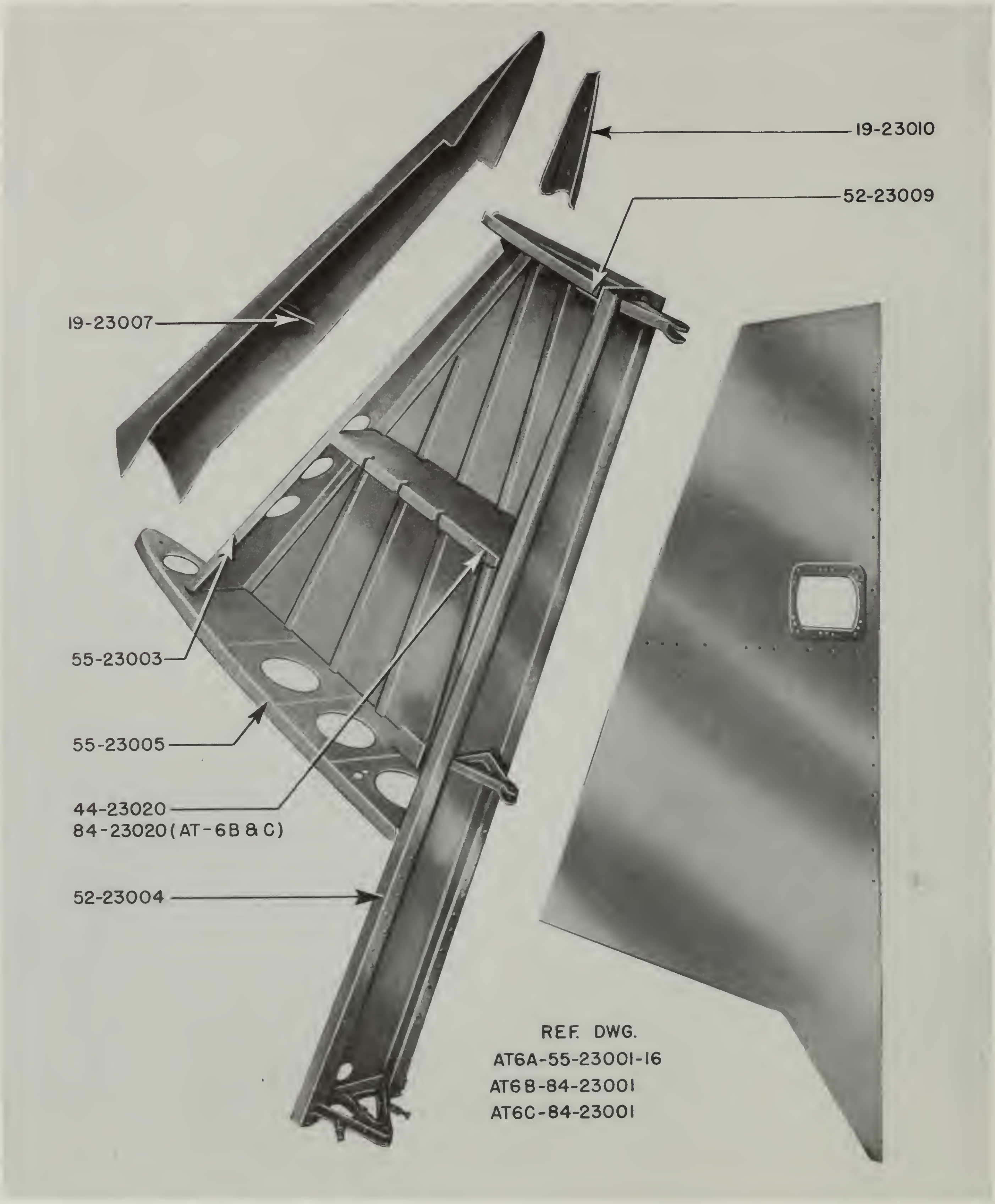


Figure 8—Vertical Stabilizer Rib and Spar Part Numbers

NOTES:
SPANWISE LINES INDICATE STRINGERS; CHORDWISE LINES INDICATE STATION LOCATIONS.

* 1/8-INCH TRIMMED FROM BULB

ITEM NO.	STRINGER	ITEM NO.	STRINGER	ITEM NO	STRINGER
2	C123LT	9	C265T	16	ROLLED .040 SHEET
3	C148T	10	C266T	17	OUTER WING MAIN SPAR
4	TWO C148T	11	C274T	18	WING FLAP SPAR
5	C204T	12	C366T	19	WING AILERON CUTOUT SPAR
6	TWO C204T	13	TWO C366T	20	CENTERSECTION REAR SPAR
7	C250T	14	K77A	21	CENTERSECTION FRONT SPAR
8	TWO C250T	15	TWO K77A	22	C123T & C140T

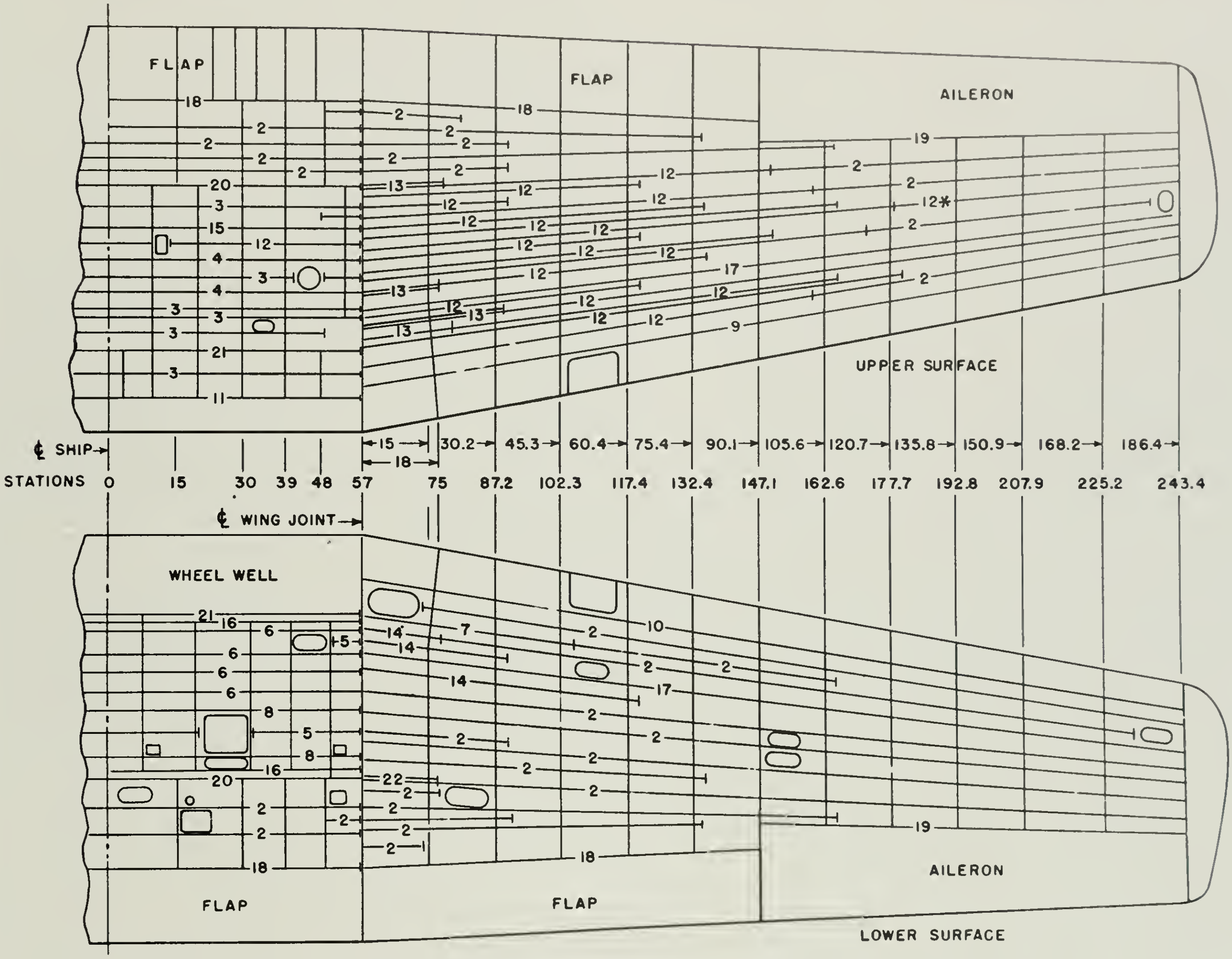


Figure 9—AT-6A Wing Stringer Arrangement and Stations

6. STRINGER TYPE C107LT-20.

Type C107LT-20 stringer is the only stringer used in the construction of the horizontal and vertical stabilizers on the AT-6A Series Airplanes. This type stringer is not used in the wing. The stringer is formed of rolled 24ST alclad sheet, .020 inch thick, having no equivalent Alcoa die number. If the damage is slight in nature, a C107LT-20 splice member, 6-1/2 inches long, will suffice *(see Figure 16)*. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut midway between two skin rivets at each side of the damage. For the splice member, cut a length of C107LT-20 stringer equal to 6-1/2 inches plus the length of the damage. Skin rivets at the end of the splice must have at least 3/8-inch edge distance and this may necessitate making the splice member slightly longer than 6-1/2 inches plus the damage length. If the prepared C107LT-20 stringer material is not available, bend up an angle locally from .020 inch thick 24ST alclad sheet material to serve as the splice member. Drill out the affected skin rivets at each side of the damage and clamp the stringer splice member to the damaged stringer. With a No. 40 (.098) drill, drill five holes at 5/8-inch on centers through the stringer upstanding flange and splice member at each side of the damage. With a drill equal to the size of the original skin rivet holes, drill through the splice member, using the existing rivet holes as a guide. If the skin rivets are spaced at 1-1/4 inches on centers or greater, double the number of existing skin rivets at the splice. Remove the splice member and burr the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice member. Clamp the splice member to the damaged stringer. At each side of the damage, drive the five AN442-AD3 rivets in the upstanding flange and rivet the splice member through the skin with the same size and type rivets previously removed from the skin *(see Figure 16)*.

7. STRINGER TYPE C123LT.

The Type C123LT stringer is used extensively in the construction of the wing, both in the upper and lower surface. The stringer is formed of rolled 24ST alclad sheet, .040 inch thick. The stringer has no equivalent Alcoa number. For damaged stringers in the lower surface of the wing, if the damage is slight in nature, a Type C203T extruded 24ST aluminum alloy splice member 7-1/2 inches in length will suffice *(see Figure 17)*. It is to be noted that, if the stringer is in the upper surface of the wing, the splice member may be shortened to 6 inches with four splice rivets and two skin rivets on each side of the damage. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut midway between two skin rivets at each side of damage. Skin rivets at the end of the splice

NOTES:
SPANWISE LINES INDICATE STRINGERS; CHORDWISE LINES INDICATE STATIONS
* 1/8-INCH MILLED FROM STRINGER BULB

ITEM NO.	STRINGER	ITEM NO.	STRINGER	ITEM NO.	STRINGER
2	C123LT	9	C123LT & C203T	16	TWO K77A
3	C148T	10	C265T	17	ROLLED .040 ALCLAD
4	TWO C148T	11	C266T	18	OUTER WING MAIN SPAR
5	C180T	12	C274T	19	WING FLAP SPAR
6	C204T	13	C366T	20	WING AILERON CUTOUT SPAR
7	TWO C204T	14	TWO C366T	21	CENTERSECTION REAR SPAR
8	TWO C250T	15	K77A	22	CENTERSECTION FRONT SPAR

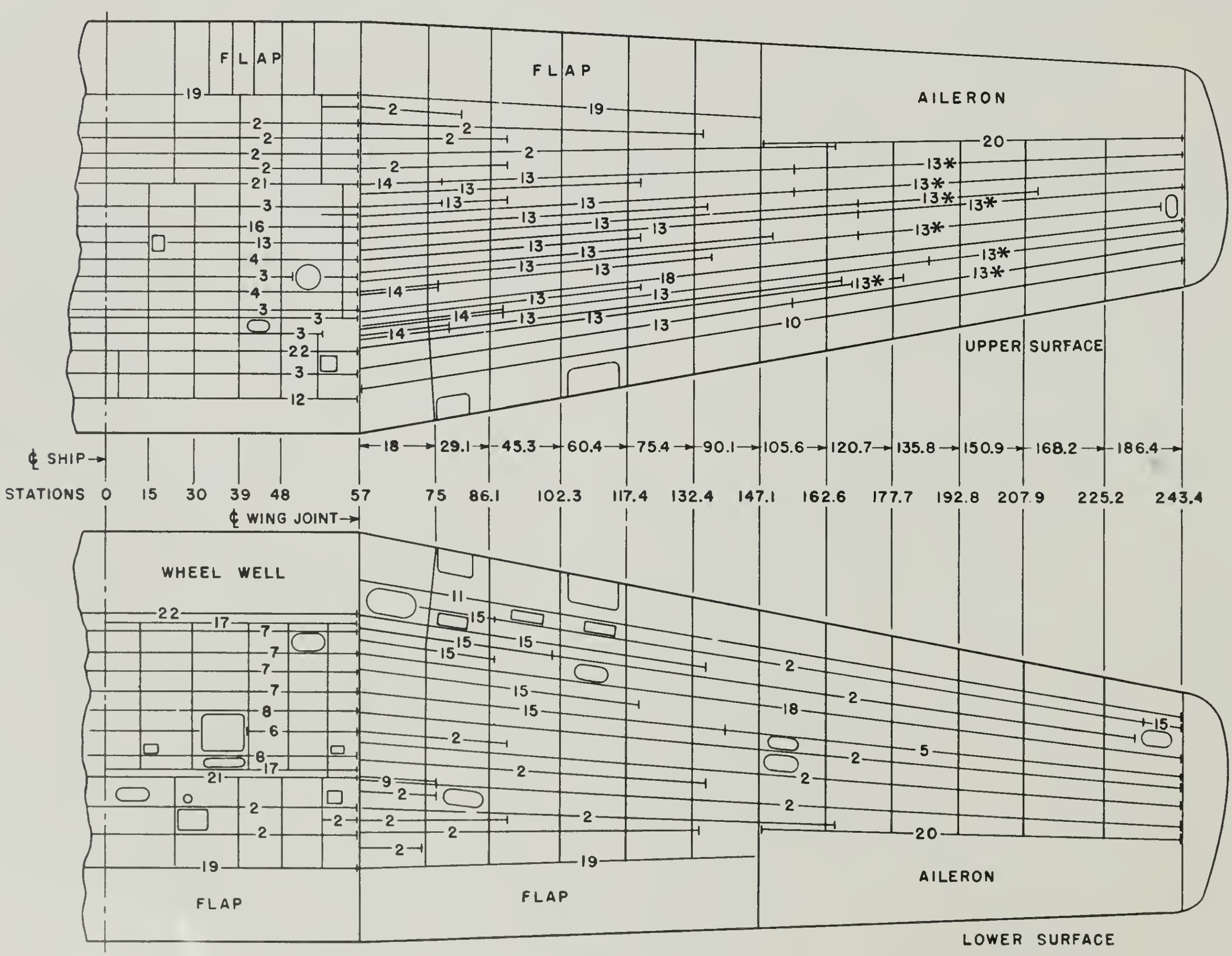


Figure 10—AT-6B and AT-6C Wing Stringer Arrangement and Stations

must have at least 3/8-inch edge distance and this may necessitate making the splice member slightly longer than that noted. If the extruded C203T splice member is not available, an .081 inch thick 24ST alclad sheet bent up to a similar shape will satisfactorily serve as the splice member. Clamp the stringer splice member to the damaged stringer. With a No. 21 (.159) drill, at each side of the damage drill the required holes at 3/4-inch on centers through the upstanding flange of the stringer and splice member. With a drill equal to the size of the skin rivet holes in the location, drill holes through the skin and splice member at the same spacing as existing skin rivets. Remove the splice member and burr the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice member and reclamp the member to the proper position on the damaged stringer. At each side of the damage, drive the AN442-AD5 rivets in the upstanding flange and rivet the splice member skin flange with the same size and type rivets employed to secure the stringer to the skin *(see Figure 17)*.

NOTE SPANWISE LINES INDICATE STRINGERS. CHORDWISE LINES INDICATE STATION LOCATIONS. DOTTED LINES INDICATE SKIN PANELS.

ITEM NOS.	STRINGER
2	TWO C250T
3	C250T-C366T
4	C265T
5	C366T
6	TWO C366T
7	MAIN SPAR

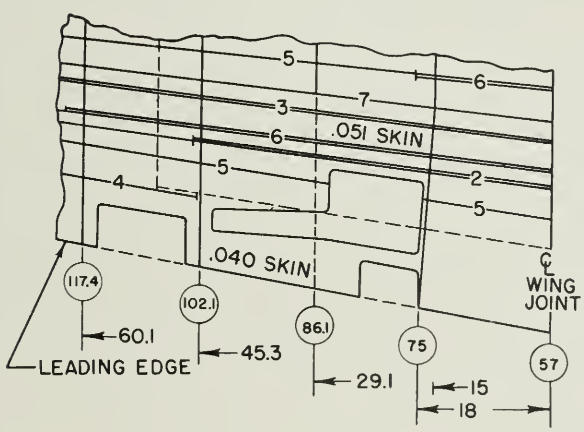
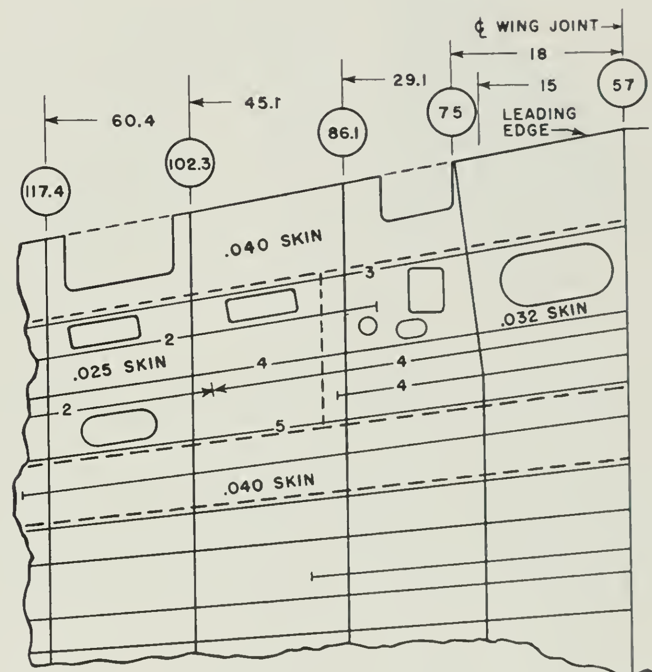


Figure 11—AT-6B and AT-6C Right Outer Wing Upper Surface Stringer Details

8. STRINGER TYPE C148T.

The Type C148T stringer is employed only in the upper surface of the wing centersection. The stringer is formed of extruded 24ST aluminum alloy material, and it has the equivalent Alcoa die number of K734JJ. If the damage is slight, the length of the C148T splice member need not exceed 10-1/4 inches *(see Figure 18)*. If the damage is extensive, cut out the damaged material, locating the cut midway between two skin rivets on each side of the damage. Cut a C148T splice member equal to 10-1/4 inches plus the length of the damaged area. Cut a length of C148T as a filler to match the removed material. Clamp the C148T splice member and the C148T filler to the proper positions on the damaged stringer; and with a No. 11 (.191) drill, drill eight rivet holes in the upstanding flanges at each side of the damage at 5/8-inch on centers. Using a drill of the same size as the existing skin rivet holes in the stringer, drill rivet locations through the



NOTE:

SPANWISE LINES INDICATE STRINGERS. CHOROWISE LINES INDICATE STATION LOCATIONS. DOTTED LINES INDICATE SKIN PANELS.

ITEM NOS.	STRINGERS
2	C123LT
3	C266T
4	K77A
5	MAIN SPAR

Figure 12—AT-6B and AT-6C Right Outer Wing Lower Surface Stringer Details

skin and splice member at the same spacing as existing skin rivets. Remove the splice member and the filler and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice member and the filler, and again clamp the members to the proper positions on the damaged stringer. At each side

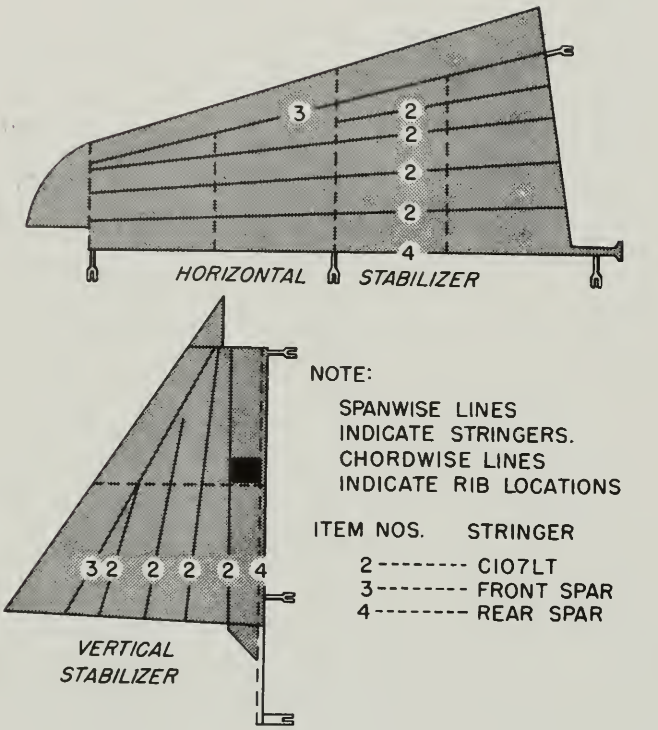


Figure 13—AT-6A Stabilizer Stringer Arrangement

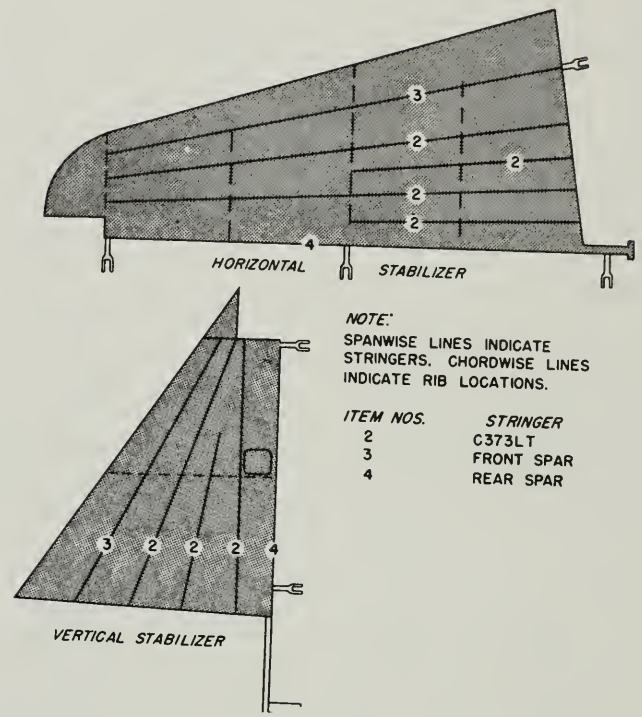


Figure 14—AT-6B and AT-6C Stabilizer Stringer Arrangement

of the damage, drive eight AN442-AD6 rivets through the upstanding flanges. Secure the skin flange of the splice member with the same size and type rivets as employed on the damaged stringer *(see Figure 18)*. Rivet the filler through the existing skin rivet holes *(see Figure 18)*.

9. DOUBLED C148T TYPE STRINGERS.

In several locations in the upper surface of the centersection, Type C148T right-angled stringers are placed together so that they form a T-shaped combination. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut midway between two skin rivets on each side of the damage. Cut sections of C148T extruded members as fillers to match the removed material. For the splice members, cut two lengths of C148T, each 10-1/4 inches long plus the length of the damage *(see Figure 19)*. Skin rivets at the ends of the splice must have at least 3/8-inch edge distance and this may necessitate making the splice members slightly longer than that noted. Chamfer the outside corners of the splice members to fit the inside radii of the damaged stringers. Drill out the affected skin rivets at each side of the damage and clamp the splice members and fillers to the damaged stringers. With a No. 11 (.191) drill, drill eight holes through the upstanding flanges of the members at a spacing of 5/8-inch on centers. With a drill equal to the size of the existing skin rivet holes, drill the required holes through the skin, stringer, and splice members. Remove the splice members, and burr the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice members and fillers. Clamp the splice members to the proper positions on the damaged stringers. At each side of the damage, insert and drive eight AN-442-AD6 rivets in the upstanding flanges of the stringer splice members. Rivet the filler to the splice members through the existing skin rivet holes. Using rivets of the same type and size as previously removed, insert and drive new rivets through the skin *(see Figure 19)*.

STRINGER TYPE	USED ON AIRCRAFT	LOCATION	APPROXIMATE TOTAL LENGTH USED PER AIRPLANE
C107LT	AT-6A	STABILIZERS	35 FEET
C123LT	ALL	WING CENTERSECTION AND OUTER PANEL	350 FEET
C148T	ALL	WING CENTERSECTION UPPER SURFACE	55 FEET
C148T (DOUBLED)	ALL	WING CENTERSECTION UPPER SURFACE	20 FEET
C180T	AT-6B AT-6C	WING OUTER PANEL LOWER SURFACE	20 FEET
C204T	ALL	WING CENTERSECTION LOWER SURFACE	7 FEET
C204T (DOUBLED)	ALL	WING CENTERSECTION LOWER SURFACE	40 FEET
C250T	AT-6A	OUTER WING LOWER SURFACE	10 FEET
C250T (DOUBLED)	ALL	WING CENTERSECTION LOWER SURFACE (WING OUTER PANEL LOWER SURFACE AT-6C ONLY)	25 FEET
C250T - C366T (COMBINATION)	AT-6B AT-6C	RIGHT OUTER WING UPPER SURFACE	10 FEET
C265T	ALL	OUTER WING UPPER SURFACE	35 FEET
C266T	ALL	OUTER WING LOWER SURFACE	35 FEET
C274T	ALL	WING CENTERSECTION UPPER SURFACE	10 FEET
C366T	ALL	WING UPPER SURFACE	275 FEET
C366T (DOUBLED)	AT-6B AT-6C	RIGHT OUTER WING UPPER SURFACE (ONLY REPAIRABLE LOCATION)	10 FEET
C373LT	AT-6B AT-6C	STABILIZERS	50 FEET
K77A	ALL	OUTER WING LOWER SURFACE	50 FEET
K77A (DOUBLED)	ALL	WING CENTERSECTION UPPER SURFACE	10 FEET

Figure 15—Summation of Stringer Types Used in Fixed Surfaces

10. STRINGER TYPE C180T.

The Type C180T stringer is used in one location in lower surface of the AT-6B and AT-6C outer wing. The stringer is a bulb angle of extruded 24ST aluminum alloy material, having the Alcoa type designation of K14280. If the stringer is only slightly damaged, a splice member 9 inches long will suffice *(see Figure 20)*. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Prepare a filler of C180T extrusion to match the removed material. Cut a section of C180T equal to the length of the damage plus 9 inches. Clamp the splice member and the filler to the proper position on the damaged stringer. At each side of the damage, center punch six rivet locations in the stringer upstanding flange at 3/4-inch intervals. With a No. 21 (.159) drill, drill through the center-punched holes. Center punch new rivet locations on the skin, using the same spacing as the existing rivets. With a drill equal to the size of the existing rivet holes in the skin, drill the center-punched holes in the skin. Remove the splice member and burr the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice member and again clamp the member to the damaged stringer. At each side of the damage, insert and drive six AN442-AD5 rivets in the stringer upstanding flange. Rivet the splice member flange and filler to the skin with the same type and size rivets employed in the damaged stringer *(see Figure 20)*.

11. STRINGER TYPE C204T.

The Type C204T stringer is employed only in the lower surface of the centersection. The stringer is formed of extruded 24ST aluminum alloy and has the equivalent Alcoa die number K14654. If the damage is of a minor nature, a 10-1/4 inch length of Type C204T extrusion will be sufficient for the splice member *(see Figure 21)*. If the damage is extensive in nature, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Prepare a filler section of C204T to match the removed material. For the splice member, cut a C204T extruded section equal to the length of the damage plus 10-1/4 inches. Clamp the splice member to the damaged stringer and center punch eight rivet locations at each side of the damage at 5/8-inch spacing. With a No. 21 (.159) drill, drill the eight rivet locations at each side of the damage. With a drill equal to the size of the existing rivet

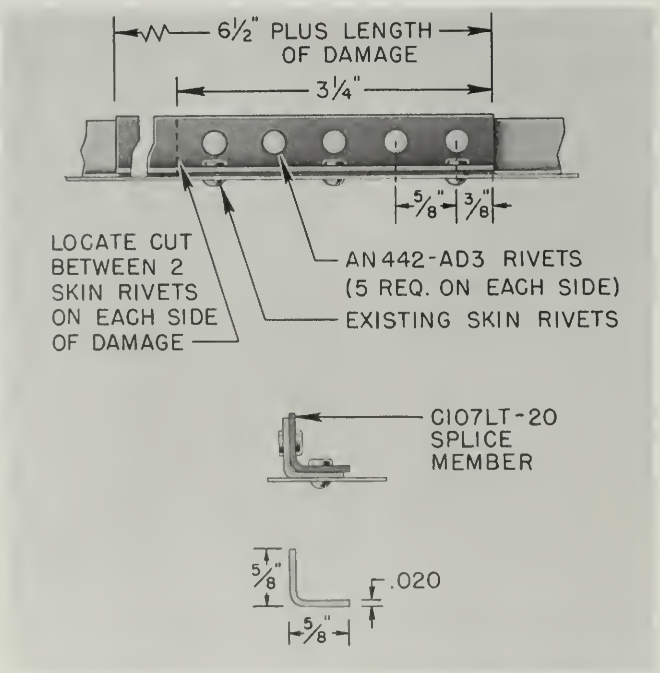


Figure 16--Splice for Stringer Type C107LT-20

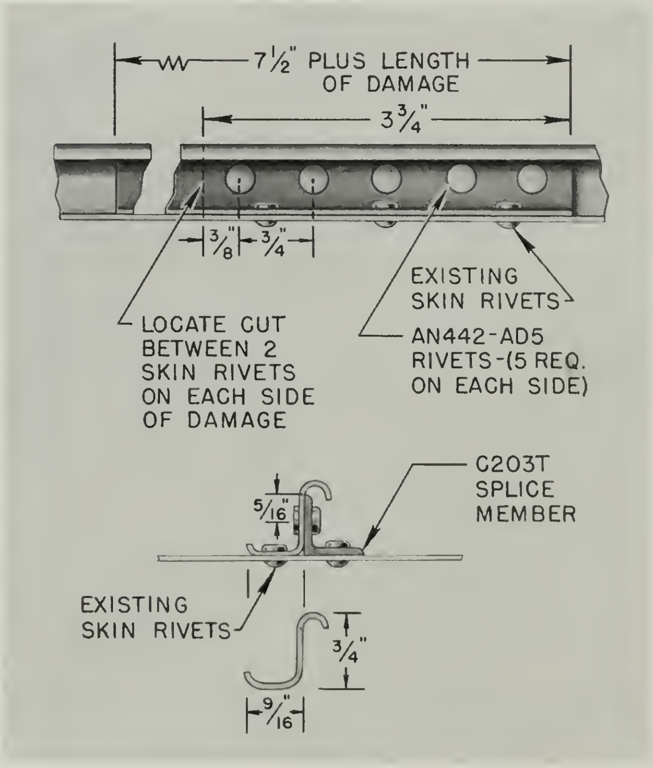


Figure 17--Splice for Stringer Type C123LT

holes in the stringer, drill new rivet locations in the splice member and filler through the skin. Remove the splice member, burr the rivet holes and apply one coat of zinc chromate primer to all surfaces of the splice member. Again clamp the splice member to the damaged stringer and insert and drive eight AN442-AD5 rivets on each side of the damage. Rivet the filler to the splice member through the existing skin holes. Using the same size and type rivets as used to secure the damaged stringer, rivet the splice member to the skin *(see Figure 21)*.

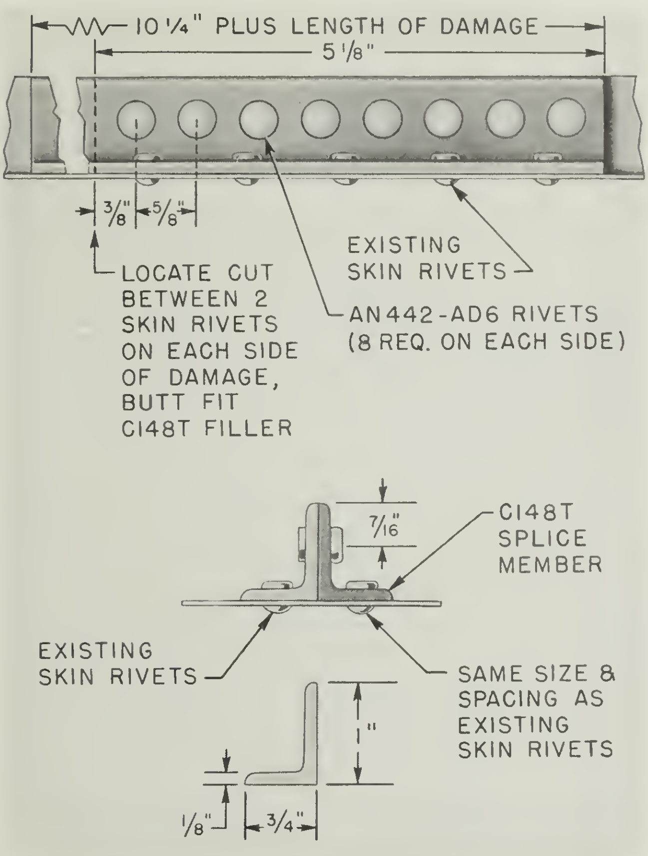


Figure 18—Splice for Stringer Type C148T

12. DOUBLED C204T TYPE STRINGERS.

The C204T doubled stringers are employed in the construction of the lower surface of the centersection. If the damage is extensive, cut out the damaged material, locating the splice midway between two skin rivets at each side of the damage. Prepare lengths of C204T sections as fillers to match the removed material. For the splice members, cut two lengths of C204T sections each equal to the length of the damage plus 10-1/4 inches *(see Figure 22)*. It is to be noted that the skin rivets at the ends of the splice must have at least 3/8-inch edge distance, and this may necessitate making the splice members slightly longer than that noted. Chamfer the outside corners of the splice members to fit the inside corners of the stringers. Drill out the affected skin rivets at each side of the damage, and clamp the fillers and the splice members to the damaged stringers. At each side of the damage, center punch eight rivet locations at 5/8-inch on centers in the upstanding flanges. With a No. 21 (.159) drill, drill the eight center-punched rivet locations at each side of the damage. With a drill equal to the size of the existing skin rivet holes, drill the splice members and fillers through the skin. Remove the splice members, and burr the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice members and reclamp the members and fillers to the

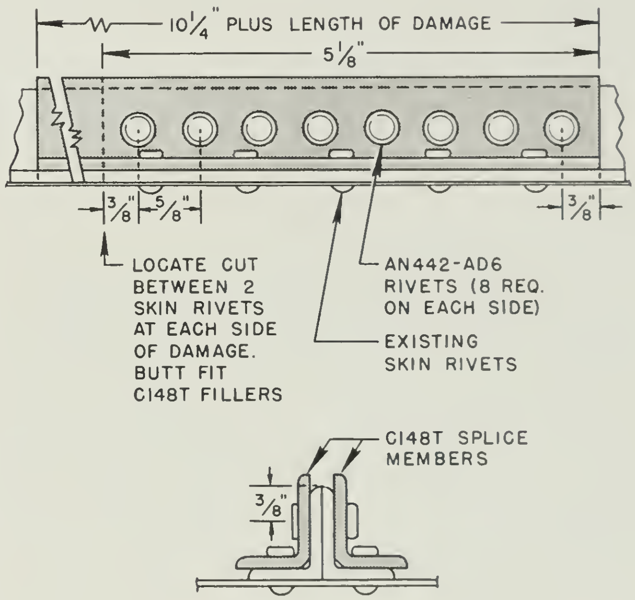


Figure 19—Splice for Doubled C148T Type Stringers

damaged stringers. At each side of the damage, insert and drive eight AN442-AD5 rivets in the upstanding flanges. Rivet the fillers to the splice members as required. Using the same type and size of rivets as originally employed, rivet the splice members through the skin *(see Figure 22)*.

13. STRINGER TYPE C250T.

The Type C250T stringer is used in only one location on the lower surface of the outer wing

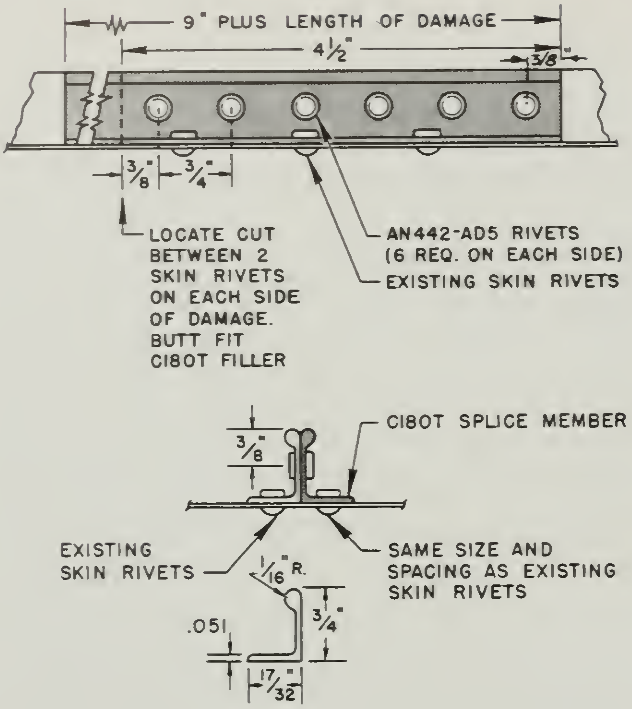


Figure 20—Splice for Stringer Type C180T

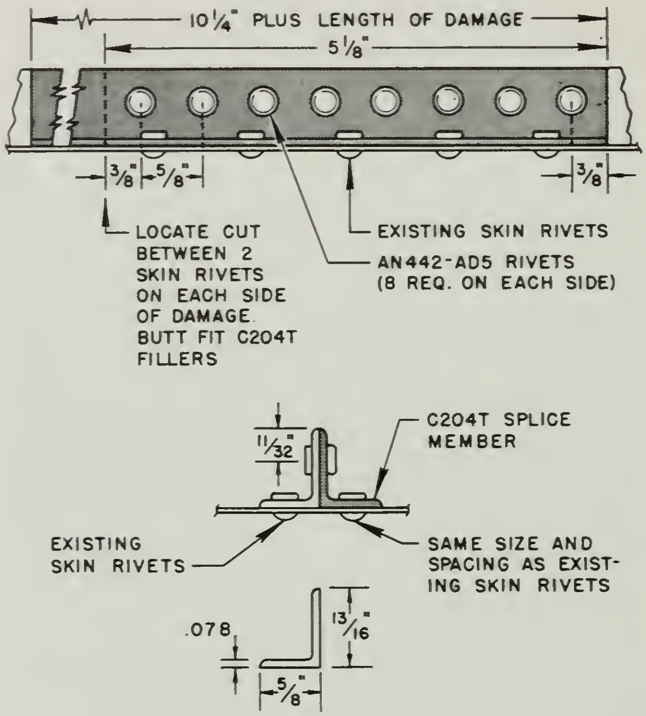


Figure 21—Splice for Stringer Type C204T

on the AT-6A Airplanes only. The stringer is formed of 24ST aluminum alloy and has the equivalent Alcoa die number K16869. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Cut a filler material of C250T extrusion to match the damaged material. For the splice members, cut a length of Type C250T extrusion equal to 10-1/4 inches plus the length of the damage, and cut a splice plate of 11/16 x 1/8 aluminum alloy also equal to the length of the damage plus 10-1/4 inches *(see Figure 23)*. Clamp the splice members and the filler to the damaged stringer. At each side of the damage, center punch eight rivet locations in the upstanding flanges; and with a No. 11 (.191) drill, drill the center-punched rivet locations. Using the same size drill as the existing skin rivet holes in the damaged stringer, drill new rivet locations through the skin and splice members at the same spacing as existing rivets. Drill the filler through the existing rivet holes. Remove the splice members and filler, and burr the rivet holes. Apply one coat of zinc chromate primer to all surfaces. Again clamp the splice members and filler to the proper position on the damaged stringer. At each side of the damage, insert and drive eight AN442-AD6 rivets in the upstanding flanges. Rivet the splice members to the filler as required. Rivet the flange of the stringer splice members to the skin with the same size and type rivets used to secure the stringer to the skin *(see Figure 23)*.

14. DOUBLED C250T TYPE STRINGERS.

The Type C250T doubled stringers are used in several locations in the lower surface of the centersection on AT-6 Series Airplanes. On the AT-6B and AT-6C only, in addition to the C250T doubled stringers employed in the centersection, the doubled stringer is also employed in the upper surface of the right outer wing just aft of the wing gun bay *(see Figure 24)*. If the damage is extensive, cut out the damaged material, locating the cut midway between two skin rivets at each side of the damage. Cut C250T filler sections to match the removed material. Cut two lengths of Type C250T extruded sections, each having a length of 10-1/4 inches plus the length of the damage *(see Figure 24)*.

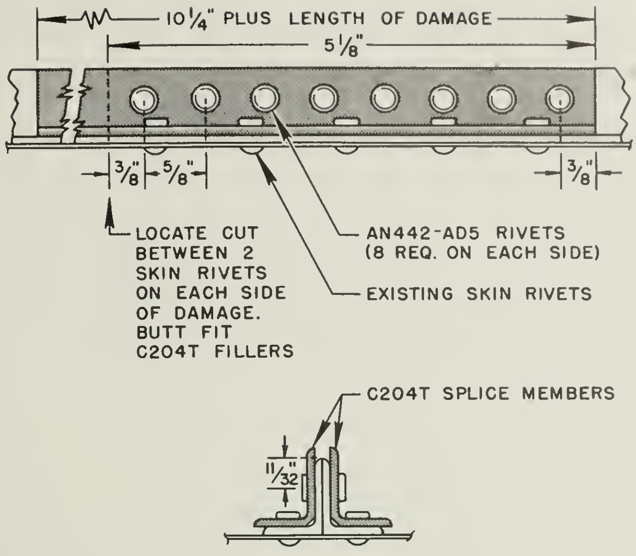


Figure 22—Splice for Doubled C204T Type Stringers

The skin rivets at the end of the splice must have at least 3/8-inch edge distance and this may necessitate making the splice slightly longer than that noted. Drill out the affected skin rivets at each side of the damage. Chamfer the outside corners of the splice members to fit the stringer radius. Clamp the splice members and the fillers to the damaged stringers. Center punch eight rivet locations in the upstanding flanges at each side of the damage at an average spacing of 5/8-inch on centers. With a No. 11 (.191) drill, drill the center-punched rivet locations. With a drill equal to the size of the existing skin rivet holes, drill through the splice members and fillers, using the existing rivet holes in the skin as a guide. Remove the splice members and fillers, and burr and clean the rivet holes. Apply one coat of zinc chromate primer to all surfaces. Again clamp the splice members and fillers to the damaged stringers. At each side of the damage in the upstanding flanges, insert and drive eight AN442-AD6 rivets. Using the same size and type rivet previously removed, rivet the splice member flanges and fillers to the skin *(see Figure 24)*.

15. COMBINATION C250T - C366T STRINGERS.

On the AT-6B and AT-6C Series Airplanes, the combination of stringer Types C250T and 366T is used in one location on the upper surface of the right outer wing just aft of the wing gun compartment *(see Figure 25)*. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Cut filler extrusions to match the removed material. For the splice members, cut a section of Type C204T extrusion equal to the length of the damage plus 11-1/2 inches; and cut a length of C250T extrusion equal to the length of the damage plus 14 inches *(see Figure 25)*. It is to be noted that the skin rivets at the ends of the splice must have at least 3/8-inch edge distance and this may necessitate making the splice members slightly longer than that noted. For a length of 6 inches on each side of the damage, trim the bulb of the damaged C366T stringer to permit fitting the C204T splice member. Drill out the affected skin rivets at each side of the damage. Chamfer the corners of the splice members to fit the inside radius of each stringer. Clamp the splice members to the proper positions on the damaged stringers. At each side of the damage in the upstanding flanges, center punch eleven rivet locations at an average spacing of 5/8-inch on centers. With a No. 21 (.159) drill, drill the center-punched rivet locations in the stringers. With a drill equal

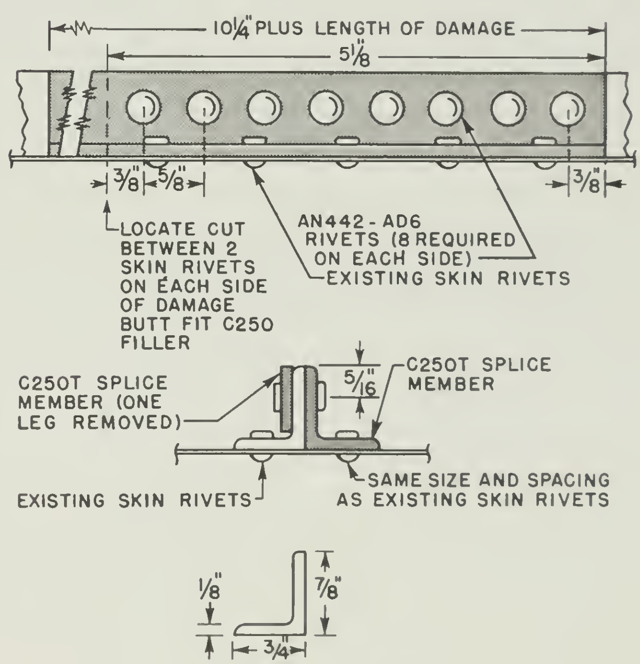


Figure 23—Splice for Stringer Type C250T

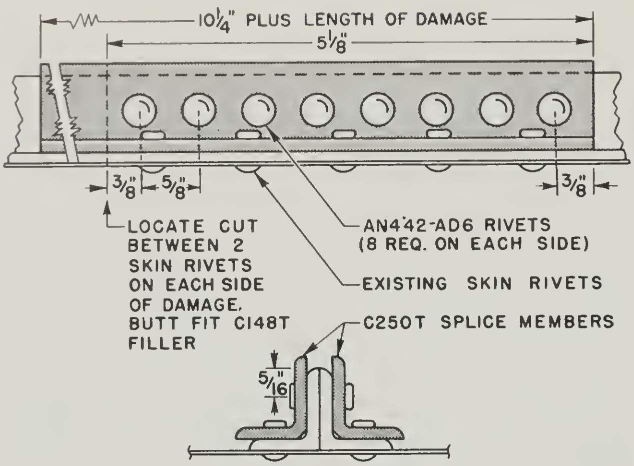


Figure 24—Splice for Doubled C250T Type Stringers

to the size of the existing skin rivet holes, drill the splice members and fillers through the skin. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice members. Again clamp the members to the damaged string-

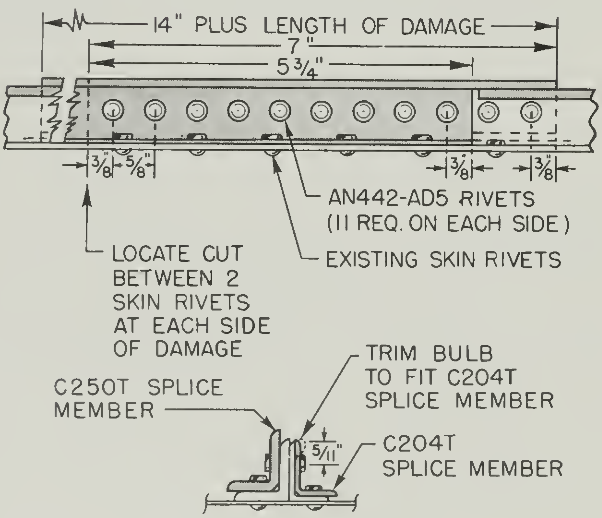


Figure 25—Splice for Combination C250T - C366T Stringers

ers. At each side of the damage in the upstanding flanges, insert and drive eleven AN442-AD5 rivets. Replace the skin rivets through the splice members and fillers with the same size and type rivets previously removed *(see Figure 25)*.

16. STRINGER TYPE C265T.

The Type C265T stringer is employed as the leading edge stringer on the upper surface of the outer wing. The stringer is formed to a 24ST aluminum alloy shape having the equivalent

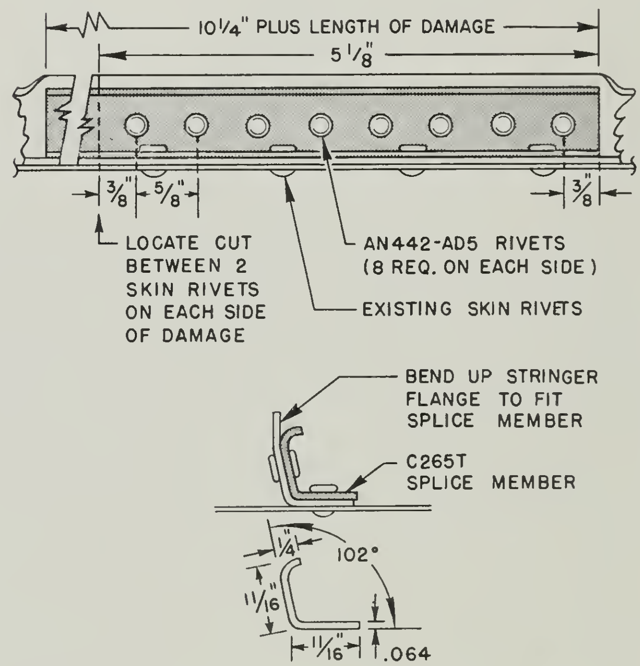


Figure 26—Splice for Stringer Type C265T

Alcoa die number L23558. If the damage is slight, a C265T splice member, 10-1/4 inches long, will suffice *(see Figure 26)*. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice member, cut a section of C265T equal to 10-1/4 inches plus the length of the damage. It is to be noted that the skin rivets at the ends of the splice member must have an edge distance of at least 3/8-inch, and this may necessitate making the splice slightly longer than that noted. At each side of the damage, drill out the affected skin rivets. In order to fit the splice member, bend up the flange of the damaged stringer *(see Figure 26)*. Clamp the splice member to the proper position on the damaged stringer. At each side of the damage in the upstanding flange of the splice member, center punch eight rivet locations at an average spacing of 5/8-inch on centers. With a No. 21 (.159) drill, drill the center-punched rivet locations. Redrill the existing skin rivet locations through the splice member with the same size drill as originally used to drill the holes. Remove the splice member and burr all the rivet holes. Apply one coat of zinc chromate to all surfaces of the splice member and reclamp the member to the damaged stringer. At each side of the damage in the upstanding flange, insert and drive eight AN442-AD5 rivets. Rivet the splice member to the skin with the same type and size rivets originally used *(see Figure 26)*.

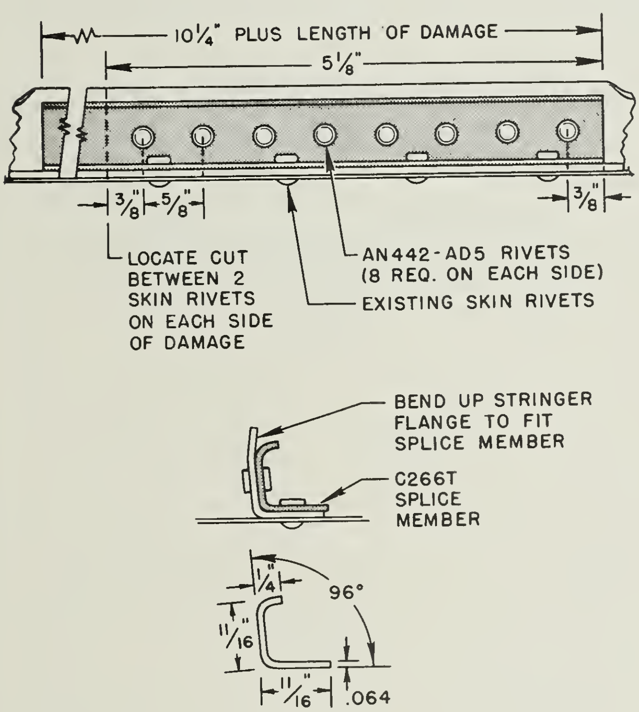


Figure 27—Splice for Stringer Type C266T

17. STRINGER TYPE C266T.

The Type C266T stringer is employed as the leading edge stringer on the lower surface of the outer wing. The stringer is formed of a 24ST aluminum alloy shape having the equivalent Alcoa die number L23558. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut between two skin rivets at each side of the damage. For the splice member, cut a section of C265T equal to 10-1/4 inches plus the length of the damage *(see Figure 27)*. The skin rivets at the ends of the splice must have an edge distance of at least 3/8-inch, and this may necessitate making the splice member slightly longer than noted. At each side of the damage, drill out the affected skin rivets. In order to fit the splice member, bend up the flange of the damaged stringer *(see Figure 27)*. Clamp the splice member to the damaged stringer. At each side of the damage in the upstanding flange of the splice member, center punch eight rivet locations at an average spacing of 5/8-inch on centers. With a No. 21 (.159) drill, drill the center-punched rivet locations. Redrill the existing skin rivet locations through the splice member with the same skin rivet hole drill as originally used. Remove the splice member and burr all the rivet holes. Apply one coat of zinc chromate to all surfaces of the splice member. Again clamp the member to the proper position on the damaged stringer. At each side of the damage in the upstanding flange, insert and drive eight AN442-AD5 rivets. Rivet the splice member through the skin with the same type and size rivets originally used through the skin *(see Figure 27)*.

18. STRINGER TYPE C274T.

The Type C274T T-angled bulb extrusion is employed as the leading edge stringer on the upper surface of the centersection. The stringer is formed to a 24ST extruded aluminum alloy shape having the equivalent Alcoa die number L23966. If the stringer is severely damaged, cut out the damaged material with a hack saw, locating the cut between two skin rivets at each side of the damage. In order to splice the stringer properly, cut two lengths of C204T extruded sections, one having a length of 9 inches plus the length of the damage and another having a length of 12-3/4 inches plus the length of the damage. Chamfer the outside corners of the splice members to fit the inside radius of the damaged stringer. Trim the bulb of the damaged stringer for a length of 4-1/2 inches on each side of the damage in order to fit the C204T splice member on the one side of the stringer. Drill out the affected skin rivets at each side of the damage and clamp the splice members to the positions indicated

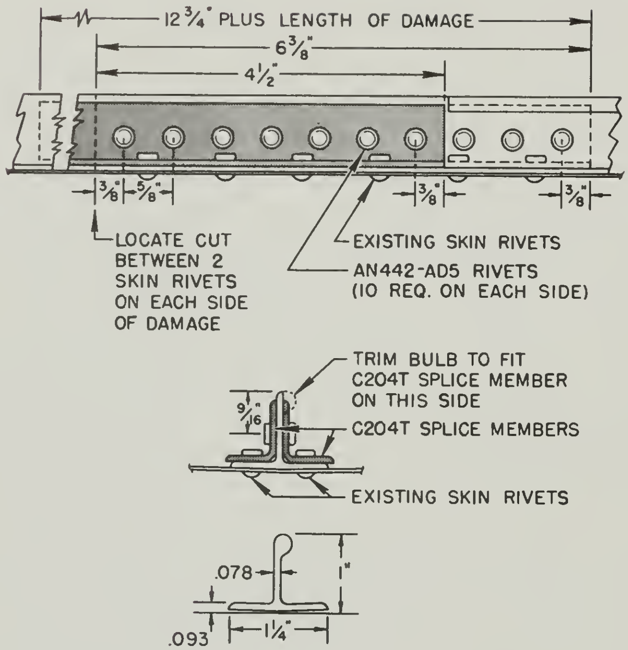


Figure 28—Splice for Stringer Type C274T

(see *Figure 28*). At each side of the damage, center punch ten rivet locations in the upstanding flanges at an average spacing of 5/8-inch on centers. With a No. 21 (.159) drill, drill the center-punched rivet locations. With a drill equal to the size of the existing holes in the skin, redrill the skin holes through the splice members. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice members. Again clamp the splice members to the damaged stringer. At each side of the damage in the upstanding flanges, insert and drive ten AN442-AD5 rivets. Secure the splice members to the skin with the same type and size rivets previously drilled out (see *Figure 28*).

19. STRINGER TYPE C366T.

The Type C366T stringer is employed extensively in the upper surface of the outer wing and in several locations in the upper surface of the centersection. The stringer is a bulb angle extrusion of 24ST aluminum alloy material having the equivalent Alcoa die number L24910. If the stringer is severely damaged, cut out the damaged material with a hack saw, locating the cut between two skin rivets at each side of the damage. Prepare a filler section of Type C366T extrusion to match the removed material. Cut a length of Type C366T extruded section equal to 9 inches plus the length of the damage (see *Figure 29*). Clamp the splice member to the damaged stringer. At each side of the damage, center punch six rivet locations in the upstanding flange at an average spacing of 3/4-inch on centers. With a No. 11 (.191) drill, drill the center-punched rivet locations. With a drill equal to the size of the existing rivet skin rivet holes, drill new rivet locations through the skin and splice member. Remove the splice member and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice member. Again clamp the splice to the damaged stringer. In the upstanding flange at each side of the damage, insert and drive six AN442-AD6 rivets. Using the same size rivets as existing skin rivets, rivet the splice member flange to the skin (see *Figure 29*).

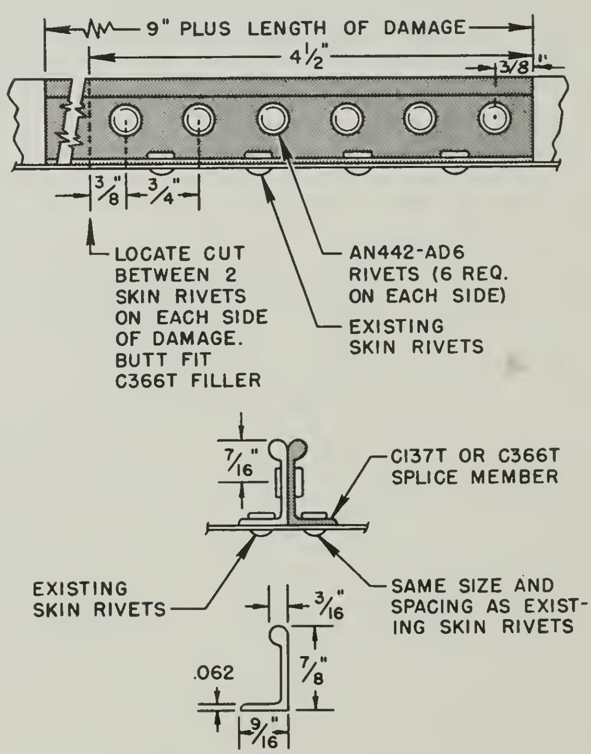


Figure 29—Splice for Stringer Type C366T

20. DOUBLED C366T TYPE STRINGERS.

The doubled combination of C366T stringers is employed in the root of the upper surface of the outer wing. It is to be noted that usually only a short length of C366T stringer is used to reinforce another C366T stringer in the root of the outer wing. If damage occurs at the root of these stringers, replace the entire length of doubled stringers, which is usually never longer than one or two bays in length, and splice the remaining C366T stringer as outlined in the preceding paragraph. Replacement of this damaged combination is recommended; but the splice outlined herein may be employed

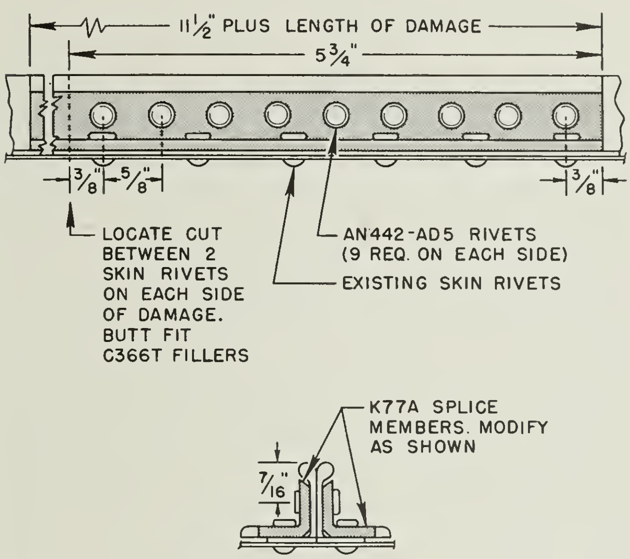


Figure 30—Splice for Doubled C366T Type Stringers

when the C366T combination is longer than two bays in length and immediate repair is necessary. This condition occurs only in one location on the upper surface of the right outer wing of the AT-6B and AT-6C Airplanes, just aft of the gun compartment *(see Figure 11)*. Proceed as follows: Cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Prepare fillers of C366T material to match the removed section of the stringer. For the splice members, cut two lengths of Type K77A extruded sections, each equal to the length of the damage plus 11-1/2 inches *(see Figure 30)*. The skin rivets at the ends of the splice must have at least 3/8-inch edge distance and this may necessitate making the splice members slightly longer than that noted. Cut and modify the splice members and chamfer the outside corners of the splice members to fit the inside radius of the stringers. Drill out the affected skin rivets. Clamp the splice members and fillers to the damaged stringers. At each side of the damage in the upstanding flanges, center punch nine rivet locations at an average spacing of 5/8-inch on centers. With a No. 21 (.159) drill, drill the center-punched rivet locations. With a drill equal to the size of the existing skin rivet holes, drill through the splice members and fillers, using the existing rivet holes as a guide. Remove the splice members and fillers and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice members and reclamp the splice members and fillers to the stringers. At each side of the damage in the upstanding flanges, insert and drive nine AN442-AD5 rivets. Replace the skin rivets with rivets of the same size and type as previously removed. Rivet the fillers to the stringers as required *(see Figure 30)*.

21. STRINGER TYPE C373LT.

The Type C373LT stringer is employed in the construction of the stabilizers of the AT-6B and AT-6C Series Airplanes. The stringer is formed of rolled alclad sheet .025 inch thick. If the damage is slight, splice members 7-1/2 inches long will suffice *(see Figure 31)*. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Bend up the two .032 inch thick splice members, each having a length equal to 7-1/2 inches plus the length of the damage *(see Figure 31)*. Drill out the affected skin rivets at each side of the damage. Clamp the splice members to the stringer. In the upstanding flange, center punch five rivet locations at an average spacing of 3/4-inch on centers. With a No. 30 (.1285) drill, drill the center-punched rivet locations. With a drill equal to the size of the existing skin rivet holes, drill through the splice members, using the existing rivet holes as a guide. Remove the splice members, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Reclamp the splice members to the damaged stringer. In the upstanding flange of the stringer at each side of the damage, insert and drive five AN442-AD4 rivets. Rivet the splice member to the skin with the same size and type rivets previously removed *(see Figure 31)*.

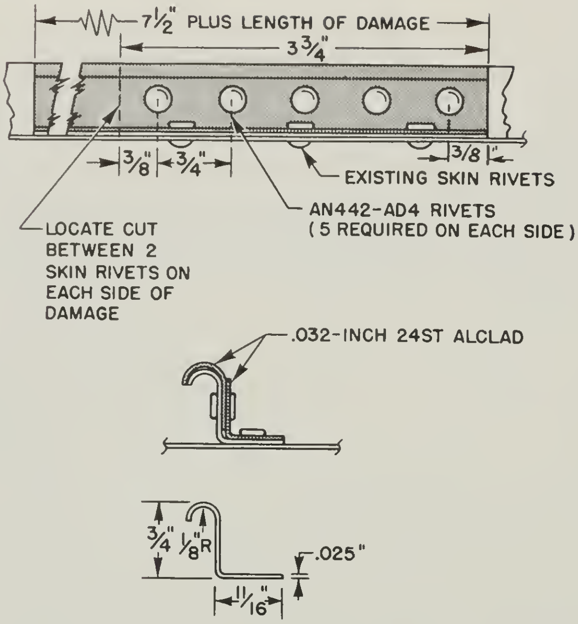


Figure 31—Splice for Stringer Type C373LT

22. STRINGER TYPE K77A.

The Type K77A Alcoa stringer is employed in several locations in the lower surface of the outer wing. The stringer is formed of 24ST aluminum alloy. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Prepare a filler of K77A extrusion to match the removed material. For the splice members, cut a length of Type K77A extrusion equal to the length of the damage plus 12-3/4 inches and bend up the same length of .040 inch thick alclad sheet *(see Figure 32)*. At each side of the damage, drill out the affected skin rivets. Clamp the splice members and fillers to the damaged stringer. At each side of the damage in the upstanding flange, center punch ten rivet locations at an average spacing of 5/8-inch on centers. With a No. 21 (.159) drill, drill the center-punched rivet locations. With a drill equal to the size of the existing skin rivet locations, redrill skin rivet holes and drill new rivet locations. Remove the splice members and burr the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the extruded splice member and to the overlapping surfaces of the alclad sheet splice member. Reclamp the splice members to the damaged stringer. At each side of the damage in the upstanding flange, insert and drive ten AN442-AD5 rivets. Replace the skin rivets with rivets of the same size and type as previously used *(see Figure 32)*.

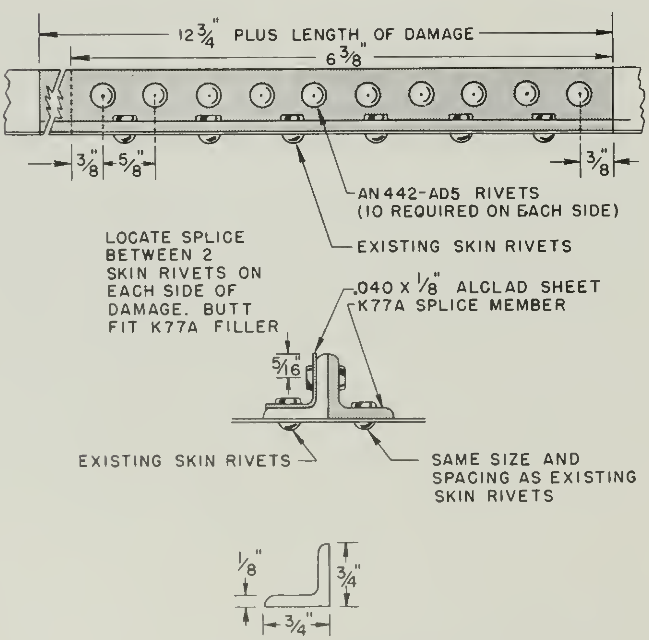


Figure 32—Splice for Stringer Type K77A

23. DOUBLED STRINGER TYPES K77A.

The Alcoa doubled K77A stringers are employed only in the upper surface of the center-section. Each stringer is formed of 24ST aluminum alloy extruded stock. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Prepare fillers of K77A extrusion to match the removed material. For the splice members, cut two lengths of K77A extrusion, each equal to the length of the damage plus 12-3/4 inches *(see Figure 33)*. Skin rivets at the ends of the splice must have at least 3/8-inch edge distance and this may necessitate making the splice members slightly longer than that noted. Drill out the affected skin rivets at each side of the damage and chamfer the outside corners of the splice members to fit the inside radii of the stringers. Clamp the splice members and fillers to the proper positions on the stringers. At each side of the damage in the upstanding flanges, center punch ten rivet locations at an average spacing of 5/8-inch on centers. Drill the center-punched rivet holes with a No. 21 (.159) drill. Using a drill equal to the size of the existing skin rivet holes, redrill the skin rivet holes through the splice members and fillers. Remove the splice members, and burr and clean the rivet holes. Reclamp the splice members

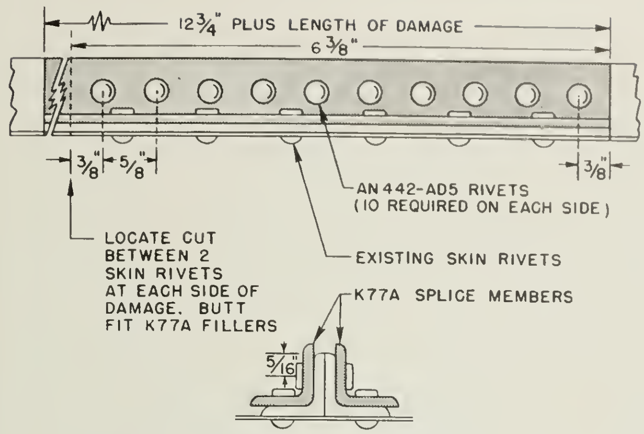


Figure 33--Splice for Doubled K77A Type Stringers

and the fillers to the damaged stringers. At each side of the damage in the upstanding flanges, insert and drive ten AN442-AD5 rivets. Replace the skin rivets through the splice members and fillers with rivets of the same type and size as previously used *(see Figure 33)*.

24. WING TRAILING EDGE.

The trailing edge of the centersection and outer wing may be spliced *(see Figure 34)*. If the damage is in the form of a crack, drill a No. 40 (.098) hole at the ends of the crack. For the splice sheets, bend up two sheets of .051 inch thick 24ST alclad to the shapes shown. Cut one of the splice sheets 9 inches long plus the length of the damage and 3 inches wide. Cut the other splice member 9 inches long plus the length of the damage and 2 inches wide. With a No. 40 (.098) drill, drill out the affected skin rivets at each side of the damage. Clamp the splice members to the trailing edge. If the damage is located in the center of the centersection where a small Z-section fairing strip is located, clamp the Z-shaped splice member over the fairing strip. With a No. 30 (.1285) drill, drill the splice members through the existing skin rivet holes. Double the number of the skin rivet holes at the splice location by drilling an additional hole between each of the existing rivet holes. With a No. 20 (.161) drill, drill twelve rivet locations through the overlap of the splice members at an average spacing of 3/4-inch on centers. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces and temporarily secure the splice members to the trailing edge with skin fasteners. Replace the skin rivets previously removed with B1227-AD4 rivets. Through the rivet holes in the overlap of the splice members, insert LS1127-5-6 Cherry blind rivets and expand the rivets with a G10 or a G15 Cherry rivet gun.

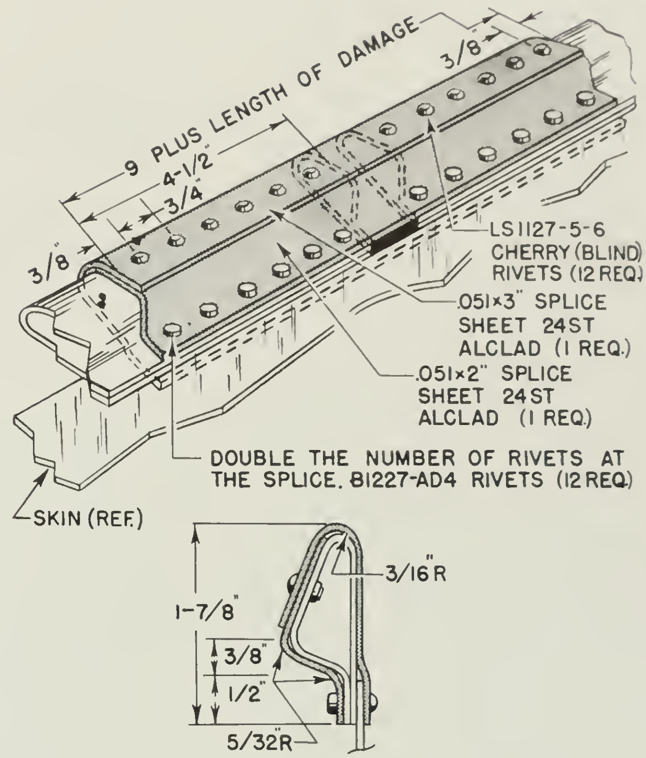


Figure 34--Splice for Wing Trailing Edge

With a pair of nippers, trim the Cherry rivet stems flush *(see Figure 34)*.

25. SPAR REPAIRS - GENERAL.

In the following paragraphs, the various splice repairs to the spars are outlined. The splice procedure and the exact location to use each splice is outlined, but it is realized that these procedures may have to be modified to suit the extent of damage encountered. Prior to making any splice to a spar, remove all load from the affected structure and hold the alignment of the structure by some jig device. Trim all damage so that there are no sharp corners to produce stress concentrations. In order to gain access properly, it may be necessary to remove a skin panel adjacent to the proposed spar repair. If this is the case, carefully drill out all skin rivets with a drill smaller than the rivet. Locate the drill in the depression on the rivet head and drill in short bursts until the rivet head twists from the shank. Drift the remainder of the rivet free. Use this method to remove all rivets from the affected skin panel. In many cases, the rib attaching angles on the spar must be removed in the affected area and then replaced after the spar repair is completed. Modify the rib attaching angles as required to permit replacement.

26. OUTER WING MAIN SPAR - ROOT TO 106 INCHES OUTBOARD.

Depending upon the extent of damage, all or part of the complete splice illustrated may be used *(see Figure 36)*. For example, if only the spar cap has been damaged, use only the spar cap part of the splice; if only the spar web is damaged, use a .064 inch thick sheet of 24ST alclad overlapping the trimmed-out round hole by 1-1/4 inches all around, secured by two rows of AN442-AD6 rivets at 7/8-inch spacing in each row. However, the procedure for a complete splice may be summed up as follows: With a spiral reamer, cut out the damaged area to a round-cornered shape or cut out the entire damaged spar section. For the splice members, cut two flat sheets of .064 inch thick 24ST alclad having a length of 10-1/2 inches plus the length of the damage and an approximate width of 8-3/4 inches. Along the length of each of the splice members, brake a 1/2-inch right-angled flange; then brake a 1-3/8 inch right-angled flange measured from the first bend. A hand brake should be used if available, or the sheet may be clamped between two blocks of wood and bent by means of hand pressure; but in either case, observe a minimum bend radius of 1/4-inch. Trim the splice members to the shape shown *(see Figure 36)*. At each side of the damage, drill out the affected skin rivets with a No. 30 (.1285) drill, taking particular care to avoid elongating the existing holes. For the web stiffeners, cut and bend four 1-1/4 inch wide strips of .064 inch thick 24ST alclad equal to the spar depth at the affected area. Bend up any additional web stiffeners to replace existing ones which may have been damaged. Securely clamp the splice members and web stiffeners to the spar. Center punch the required rivet locations as noted, and drill these locations with a No. 11 (.191) drill. It is to be noted that all rivets at the ends of the splice must have at least 3/8-inch edge distance, measured from the center-line of the hole. With a drill equal to the existing skin rivet holes, drill the splice members through the existing skin rivet holes. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice members and web stiffeners and reclamp the members to the proper positions on the spar. Into the No. 11 (.191) holes, insert and drive AN442-AD6 rivets. To prevent the sheet from creeping, rivet first at the corners and then at the intermediate positions. On each side of the damage through the upper skin, insert and drive at least ten AN426-AD rivets of the same size as previously used. On each side of the damage through the lower skin, insert and drive at least ten B1227-AD rivets of the same size as previously used *(see Figure 36)*.

27. OUTER WING MAIN SPAR - 106 TO 151 INCHES OUTBOARD OF ROOT.

If only the spar cap has been damaged, use only the spar cap part of the splice; if only the spar web is damaged, use a .040 inch thick patch of 24ST alclad overlapping the hole by

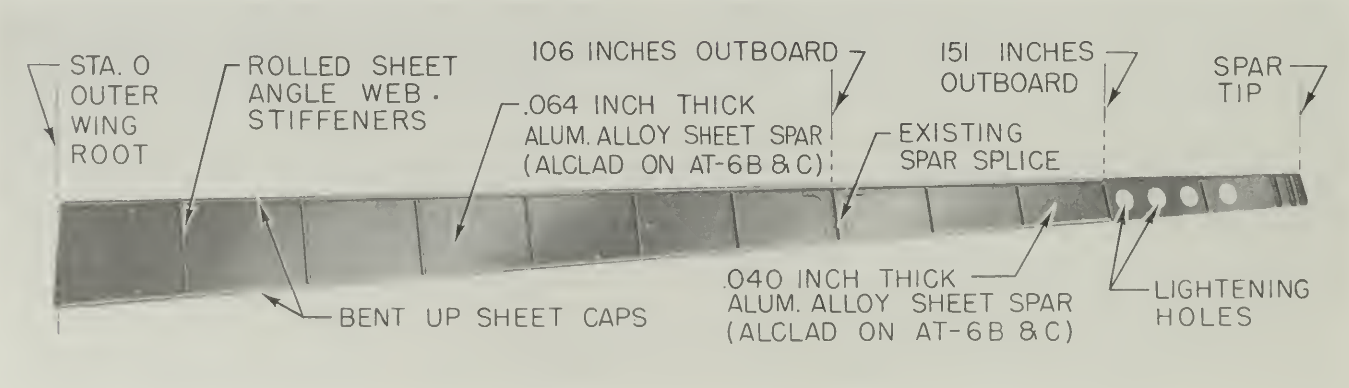


Figure 35--Outer Wing Main Spar

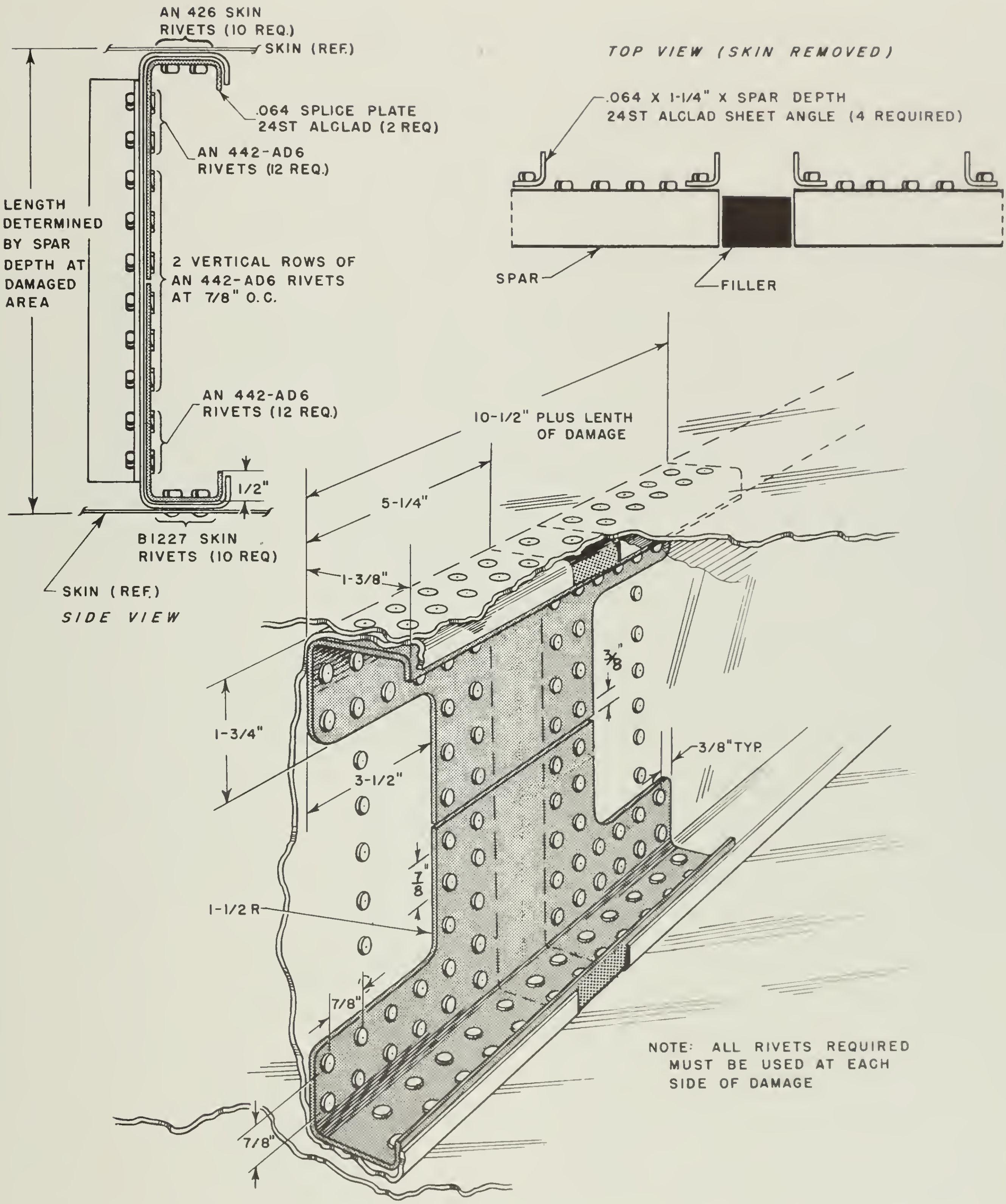


Figure 36—Outer Wing Main Spar Splice - Root to 106 Inches Outboard

1-1/4 inches all around. Secure the patch with two rows of AN442-AD5 rivets at 3/4-inch spacing in each row. However, the procedure for a complete splice may be summed up as follows *(see Figure 37)*. With a spiral reamer, cut out the damaged area to a round-cornered shape, or cut out the entire damaged spar section. For the splice members, cut two flat sheets of .040 inch thick 24ST alclad having a length of 7-1/2 inches plus the length of the damage and an approximate width of 6-1/4 inches. Along the length of each of the splice members, brake a 1/2-inch right-angled flange, and then brake a 1-3/8 inch right-angled flange measured from the first bend. A hand brake should be used if available, or the sheet may be clamped between two blocks of wood and bent by means of hand pressure; but in either case, observe a minimum bend radius of 5/32-inch. Trim the splice members to the shapes shown *(see Figure 37)*. At each side of the damage, drill out the affected skin rivets with a No. 40 (.098) drill, taking particular care to avoid elongating the existing holes. For the web stiffeners, cut and bend up four lengths of 1-3/8 inch strips of .051 inch thick 24ST alclad equal to the spar depth at the affected area. Bend up any additional web stiffeners to replace existing ones which may have been damaged. Securely clamp the splice members and web stiffeners to the proper positions on the spar. Center punch the required rivet locations as noted, and drill these locations with a No. 21 (.159) drill. With a No. 30 (.1285) drill, drill the splice members through the existing skin rivet holes. It is to be noted that all rivets at the ends of the splice members must have an edge distance of at least 3/8-inch measured from the centerline of the hole. Remove the splice members and web stiffeners and burr all rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Reclamp the members to the proper positions on the spar. Into the No. 21 holes, insert and drive AN442-AD5 rivets. To prevent the sheet from creeping, rivet first at the corners and then at the intermediate positions. On each side of the damage through the upper skin, insert and drive at least eight AN426-AD4 rivets. On each side of the damage through the lower skin, insert and drive at least eight B1227-AD4 rivets *(see Figure 37)*.

28. OUTER WING MAIN SPAR - 151 INCHES OUTBOARD TO TIP.

Damage to the outer panel main spar from 151 inches outboard to the tip of the wing may be repaired as shown *(see Figure 38)*. The splice is applicable only to areas between flanged web cutouts. Where flanged web cutouts are present in the affected area, trim the doublers around the web cutouts as required, making certain that the required number of rivets is picked up at each side of the damage. With a spiral reamer, cut out the damaged area, locating the cut between two skin rivets at each side of the damage. For the splice members, cut two flat sheets of .040 inch thick 24ST alclad, having a length of 6 inches plus the length of the damage and having an approximate width of 4-7/8 inches. It is to be noted that if the damage occurs in areas adjacent to web cutouts, the length of the splice members may have to be increased to pick up the required rivets at each side of the damage. Along the length of each of the splice members, brake a 1/4-inch right-angled flange, and then brake a 1-inch right-angled flange measured from the first bend. A hand brake should be used if available, orthe sheet may be clamped between two blocks of wood and bent by means of hand pressure; but in either case, a minimum bend radius of 5/32-inch should be observed. Trim the splice members to the shapes shown *(see Figure 38)*, or trim the members around cutouts as required. At each side of the damage, drill out the affected skin rivets with a No. 40 (.098) drill, taking particular care to avoid elongating the existing holes. For the web stiffeners, bend up two 1-3/8 inch strips of .051 inch thick 24ST alclad equal to the spar depth at the affected area. Bend up any additional required web stiffeners to replace existing ones which may have been damaged. Securely clamp the splice members and the web stiffeners to the proper positions on the spar. Center punch the required rivet locations as noted and drill these locations with a No. 21 (.159) drill. With a No. 30 (.1285) drill, drill the splice members through the existing skin rivet holes. It is to be noted that all rivets at the end of the splice must have an edge distance of 3/8-inch measured from the centerline of the hole. Remove the splice members and the web stiffeners, and burr all rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the members to the spar. Into the No. 21 (.159) holes, insert and drive AN442-AD5 rivets. To prevent the sheet from creeping, rivet first at the corners and then at the intermediate positions. On each side of the damage through the upper skin, insert and drive at least four AN426-AD4 rivets. On each side of the damage through the lower skin, insert and drive at least four B1227-AD4 rivets *(see Figure 38)*.

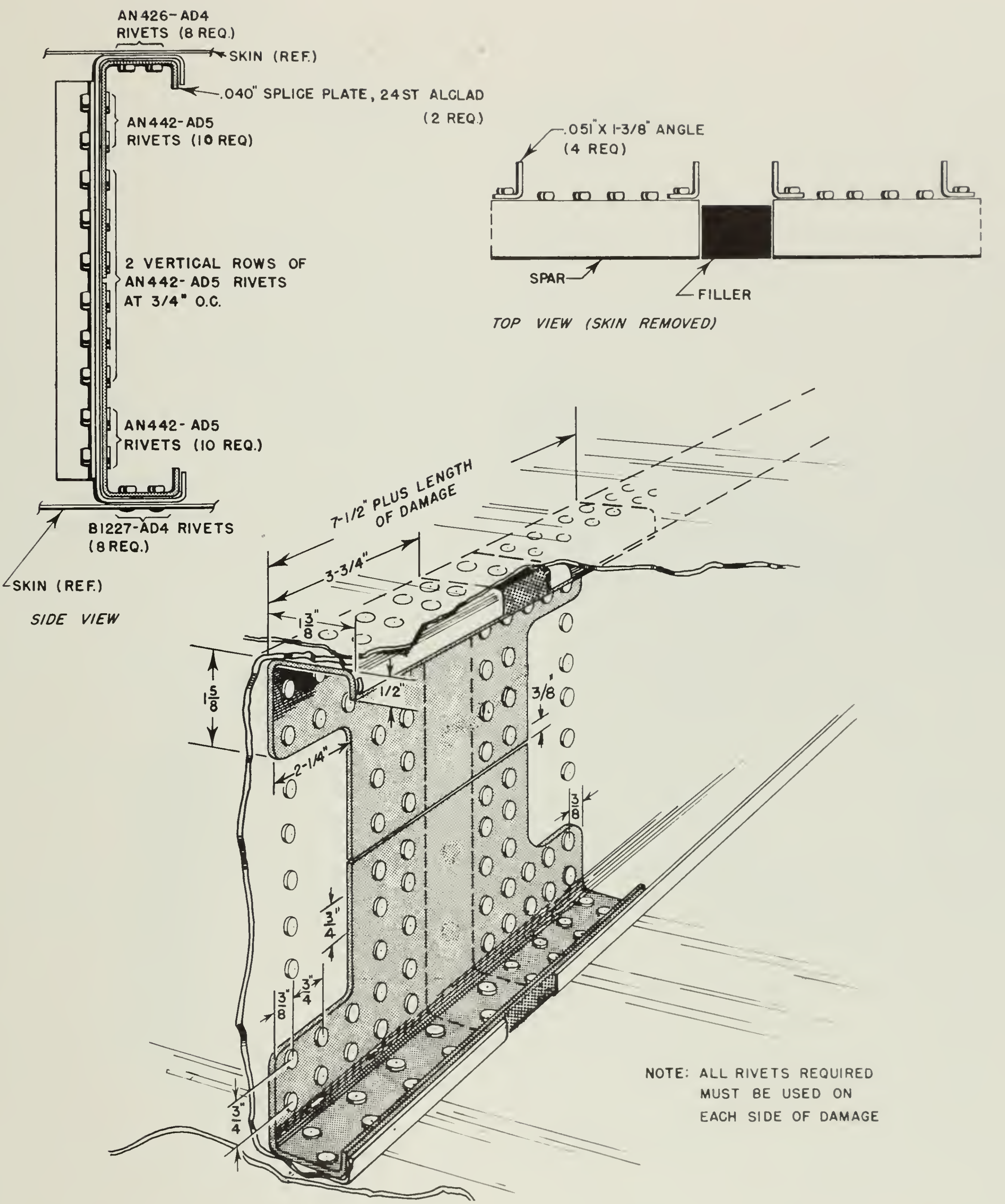


Figure 37—Outer Wing Main Spar Splice - 106 to 151 Inches Outboard of Root

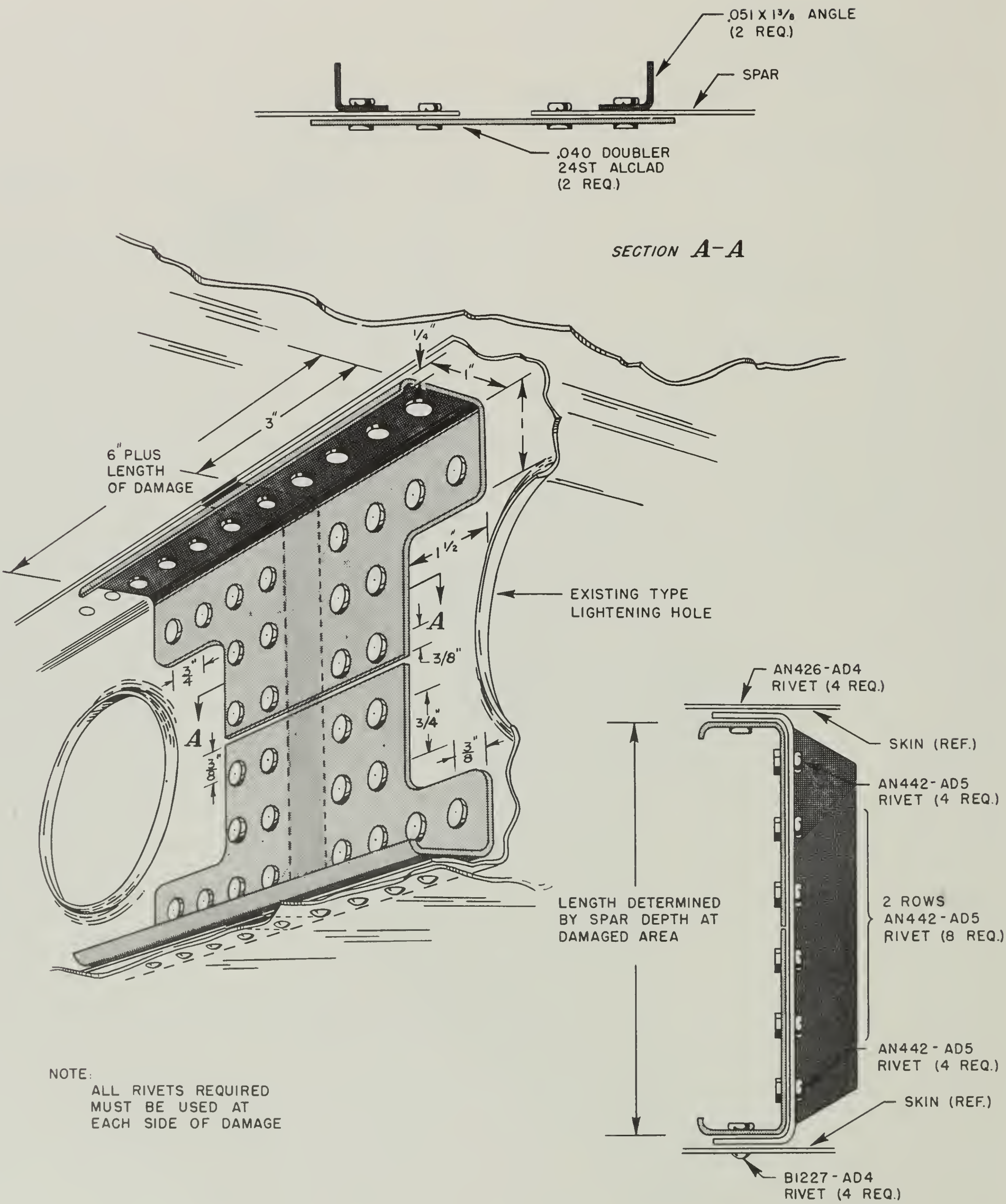


Figure 38—Outer Wing Main Spar Splice - 151 Inches Outboard to Tip

29. OUTER WING FLAP SPAR.

Damage to the outer wing flap spar (located in the outer wing trailing edge structure inboard of the aileron) may be repaired as shown *(see Figure 39)*. With a spiral reamer, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice members, cut two flat sheets of .040-inch 24ST alclad, having a length of 6-1/2 inches plus the length of the damage and an approximate width of 3-1/2 inches. It is to be noted that, if the damage occurs between two lightening holes in the spar, the length of the splice members may have to be extended to pick up the required number of rivets. Along the length of the upper splice member, brake a 77-degree, 3/4-inch flange. Along the length of the lower splice member, brake an 86-degree, 7/16-inch flange. A hand brake should be used if available, or the sheet may be clamped between two blocks of wood and bent by means of hand pressure; but in either case, a minimum bend radius of 1/8-inch should be observed. Trim the splice members so that they will fit around the lightening holes. At each side of the damage, drill out the affected skin rivets with a No. 40 (.098) drill, taking particular care to avoid elongating any existing holes. For the web stiffeners, bend up two 1-1/4 inch strips of .032 inch thick 24ST alclad equal to the spar depth at the affected area. If damage occurs to any of the extruded stiffeners employed at intervals along the spar length, remove and replace the stiffeners with similar extrusions. Securely clamp the splice members and the web stiffeners to the damaged spar. Center punch the required rivet locations as noted and drill these locations with a No. 30 (.1285) drill. With the same drill, drill the cap of the upper splice member through the existing skin rivet holes. With a No. 40 (.098) drill, drill the cap of the lower splice member through the existing skin rivet holes. It is to be noted that all rivets at the ends of the splice must have an edge distance of 3/8-inch measured from the centerline of the hole. Remove the splice members and the web stiffeners, and burr all rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the members to the proper positions on the spar. Into the spar web rivet holes, insert and drive AN442-AD4 rivets. Replace the skin rivets with B1227-AD4 rivets on the upper cap and B1227-AD3 rivets on the lower cap. To prevent the sheets from creeping, rivet first at the corners and then at the intermediate positions. This completes the repair *(see Figure 39)*.

30. OUTER WING AILERON SPAR.

Damage to the outer wing aileron spar (located in the outer wing trailing edge structure outboard of the flap) may be repaired as shown *(see Figure 40)*. With a spiral reamer, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice members, cut two flat sheets of .032 inch thick 24ST alclad, each having a length of 7-1/2 inches plus the length of the damage and an approximate width of 4 inches. It is to be noted that the skin rivets at the end of the splice must have an edge distance of 3/8-inch and this may necessitate making the splice members slightly longer than that noted. Along the length of the upper splice member, brake a 100-degree, 3/4-inch flange. Along the length of the lower splice member, brake a 93-degree, 3/4-inch flange. A hand brake should be used if available, or the sheets may be clamped between two blocks of wood and bent by means of hand pressure; but in either case, a minimum bend radius of 1/8-inch should be observed. Trim the splice members so that they will fit around the lightening holes in the damaged spar. At each side of the damage, drill out the affected skin rivets with a No. 40 (.098) drill, taking particular care to avoid elongating the existing rivet holes. For the web stiffeners, bend up two 1-1/4 inch strips of .032 inch thick 24ST alclad equal to the spar depth at the affected area. Securely clamp the splice members and the web stiffeners to the proper positions on the damaged spar. Center punch the required rivet locations as noted and drill these locations with a No. 30 (.1285) drill. With the same drill, drill the splice members through the existing skin rivet holes. Remove the splice members and the web stiffeners and burr all rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the members to the proper positions on the spar. Into the spar web rivet holes, insert and drive AN442-AD4 rivets. Replace the skin rivets with B1227-AD4 rivets *(see Figure 40)*. To prevent the sheets from creeping, rivet first at the corners and then at the intermediate positions.

31. CENTERSECTION FLAP SPAR.

Damage to the centersection flap spar (located in the centersection structure forward

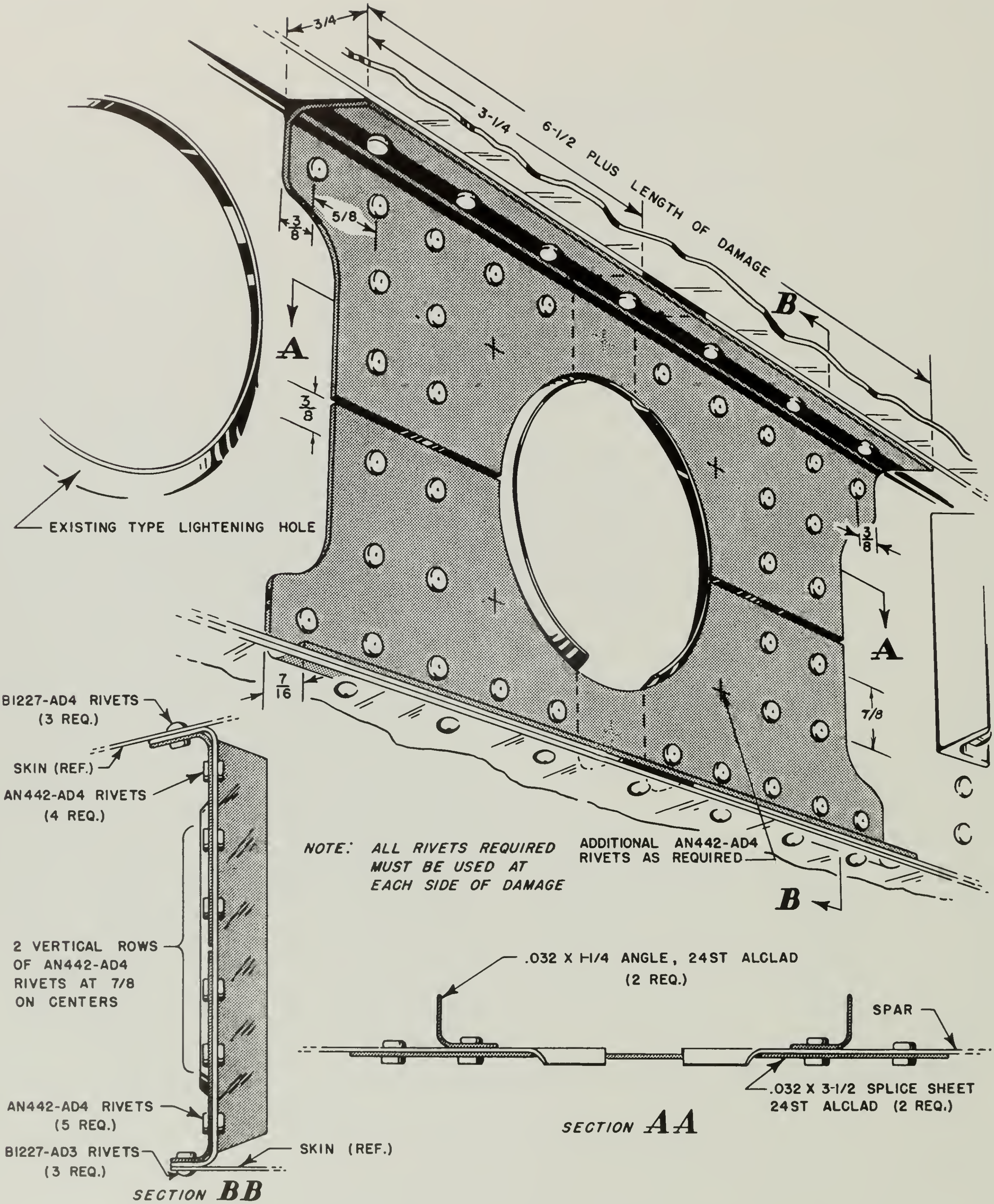


Figure 39—Outer Wing Flap Spar Splice

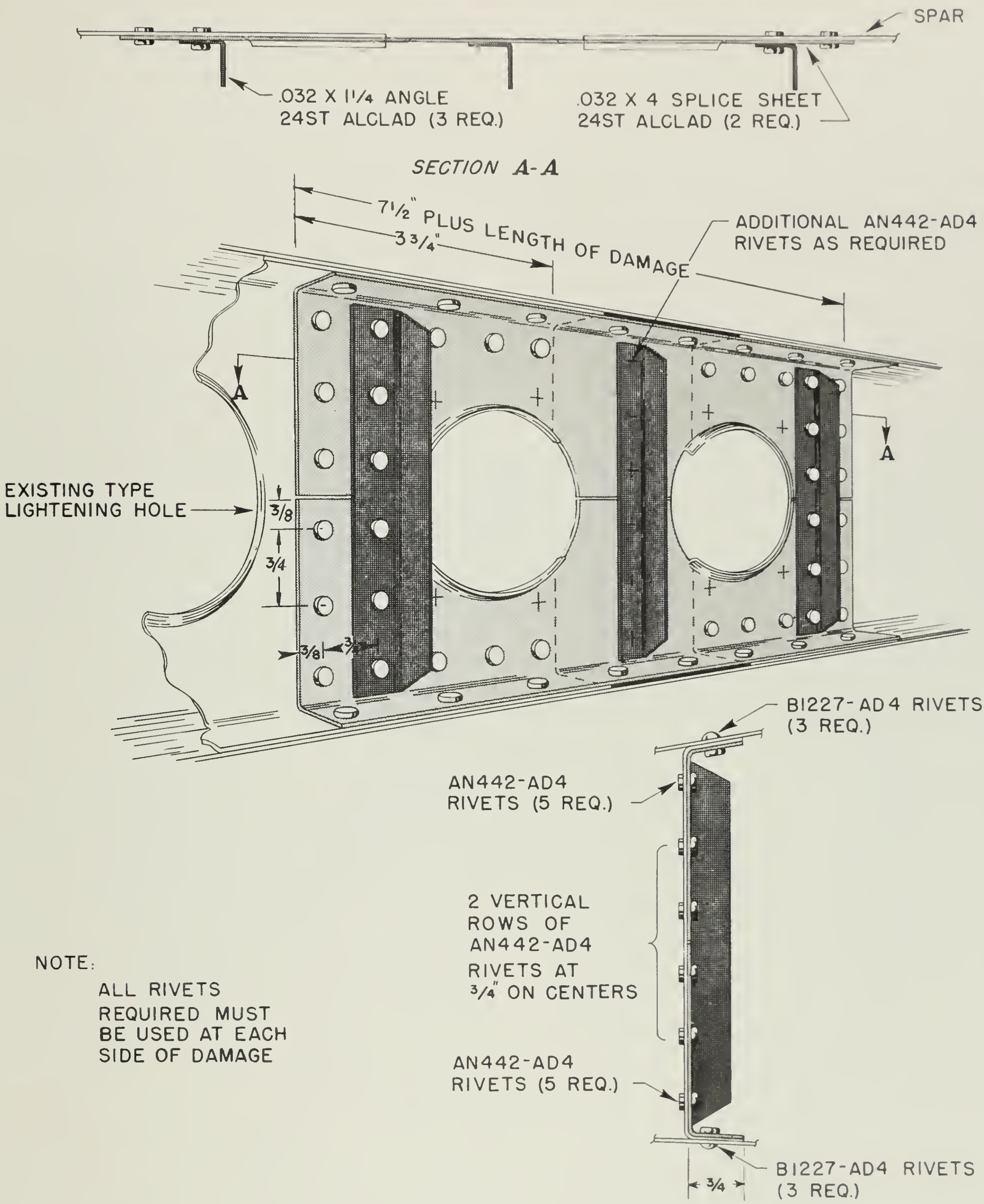


Figure 40—Outer Wing Aileron Spar Splice

of the flap) may be repaired as shown *(see Figure 41)*. The complete splice of the member is shown; but depending upon the extent of damage, all or part of the complete splice may be used. With a spiral reamer, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice members, cut two flat sheets of .032 inch thick 24ST alclad, having a width of 3-5/16 inches and sufficient length to pick up the required number of rivets at each side of the damage. In the repair illustrated, the length of the splice member is equal to the length of the damage plus 10-1/2 inches. Along the length of the upper splice member, brake a 78-degree, 3/4-inch flange. Along the length of the lower splice member, brake an 86-degree, 7/16-inch flange. A hand brake should be used if available, or the sheets may be clamped between two blocks of wood and bent by means of hand pressure. Observe a minimum bend radius of 1/8-inch. Trim the splice members so that they will fit around the lightening holes. At each side of the damage, drill out the affected skin rivets with a No. 40 (.098) drill, taking particular care to avoid elongating the existing rivet holes. For the web stiffeners, bend up three strips of .032 inch thick 24ST alclad, 1-1/4 inches wide and 5 inches long. If damage occurs to any of the extruded angles used at intervals along the spar length, remove and replace the angles with similar extrusions. Securely clamp the splice members and the web stiffeners to the spar. Lightly center punch the required rivet locations as noted and drill these locations with a No. 30 (.1285) drill. With the same drill, drill the cap of the upper splice member through the skin rivet holes. With a No. 40 (.098) drill, drill the cap of the lower splice member through the existing skin rivet holes. Remove the splice members and the web stiffeners, and burr all rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the members to the proper positions on the spar. Into the spar web rivet holes, insert and drive AN442-AD4 rivets. Replace the skin rivets with B1227-AD4 rivets in the upper cap and B1227-AD3 rivets in the lower cap *(see Figure 41)*.

32. CENTERSECTION REAR SPAR.

Damage to the centersection rear spar (located in the centersection structure just aft of the fuel tank compartments) may be repaired as illustrated *(see Figure 43)*. The complete splice of the member is illustrated; but depending upon the extent of the damage, all or part of the complete splice may be used. For example, if only the spar web is damaged, round out the hole to an oval shape; prepare an .051 inch thick 24ST alclad patch with a 1-1/4 inch overlap and secure the patch with two rows of AN442-AD5 rivets at 3/4-inch spacing all around the hole. With a spiral reamer or a hack saw, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Remove any damaged stiffeners in the area. With a No. 30 (.1285) drill, drill out the affected skin rivets at each side of the damage, taking particular care to avoid elongating any existing rivet holes. On the lower cap of the rear spar on the AT-6A Series Airplanes, drill out the rivets securing the existing B1083 stop nuts in the affected area. On the AT-6B and C Series Airplanes, the single steel nut channel assembly (NA Dwg. 84-13332) spanning the entire length of the rear spar lower cap must be removed by drilling out the securing rivets. Remove the channel and, with a hack saw, cut out the damaged section of the channel 5-1/4 inches from each side of the damage between two of the channel nuts. At each side of the damage, rerivet the two remaining sections of the nut channel to the lower spar cap, discarding the damaged section. To replace the damaged section of the nut channel, cut a length of 22GCH-048 steel channel to match the damaged section and snap in the required number of 22G1-048 nuts. (Channel and nuts may be purchased from the Boots Nut Corporation, Waterbury, Conn.) For the splice members, cut two sheets of .051 inch thick 24ST alclad having a length equal to the length of the damage plus 10-1/2 inches and a width of 7-3/16 inches. The skin rivets at the ends of the splice must have an edge distance of at least 3/8-inch and the bolts must have 1/2-inch edge distance; this may necessitate making the splice members slightly longer than that noted. Along the length of the upper splice member, brake a 90-degree, 1/2-inch flange and then brake a 99-degree, 1-3/4 inch flange measured from the first bend. Along the length of the lower splice member, brake a 90-degree, 1/2-inch flange and then brake a 92-degree, 1-3/4 inch flange measured from the first bend. A hand brake should be used if available. Observe a minimum bend radius of 3/16-inch. Trim the splice members to the shapes shown *(see Figure 43)*. For the extruded stiffeners, cut two lengths of Type C204T extrusion, each having a length of 9-5/8 inches. Cut similar lengths of the extrusion to replace any of the existing stiffeners which may have been damaged. Clamp the splice members and the web stiffeners to the proper positions on the

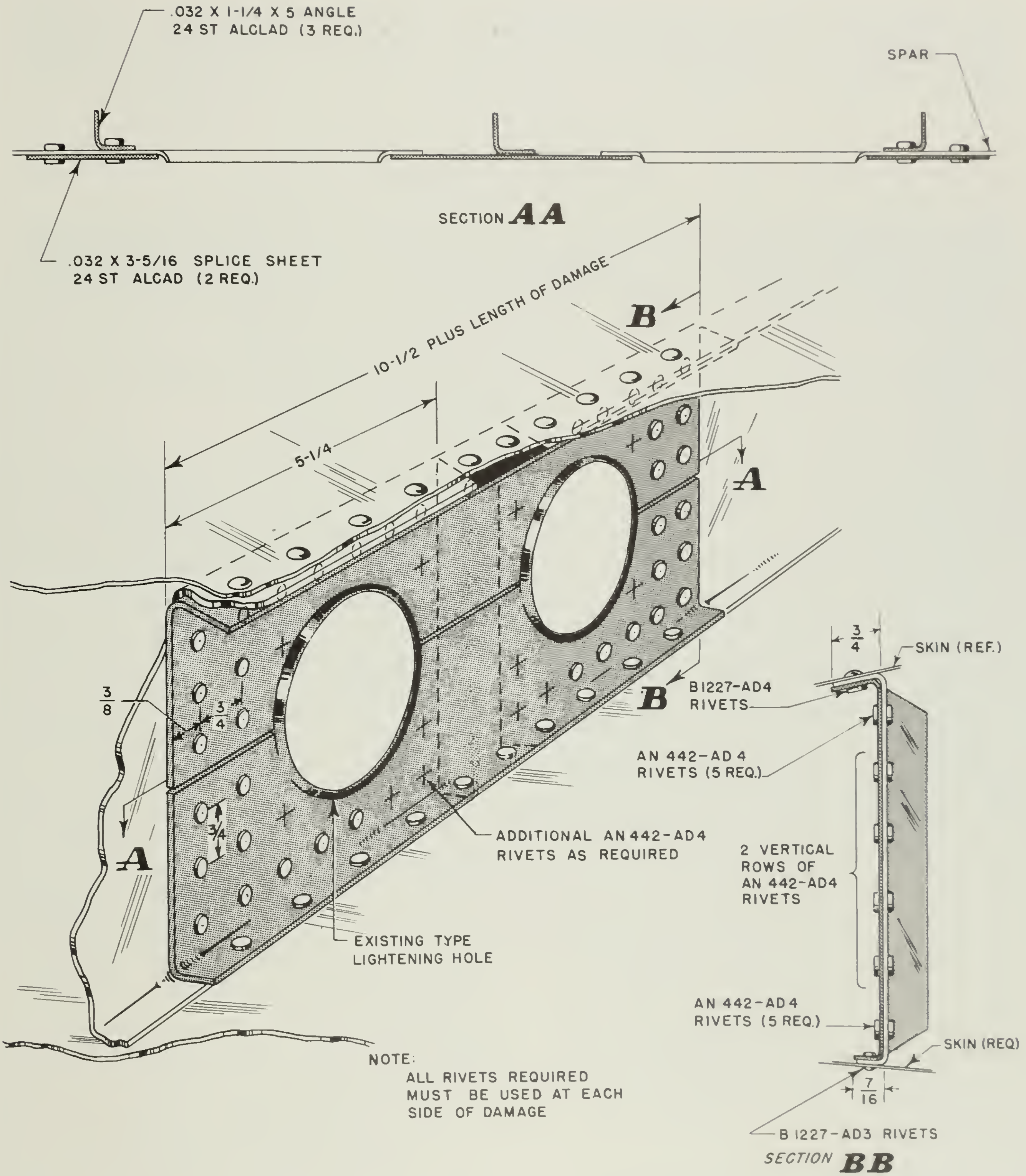


Figure 41—Centersection Flap Spar Splice

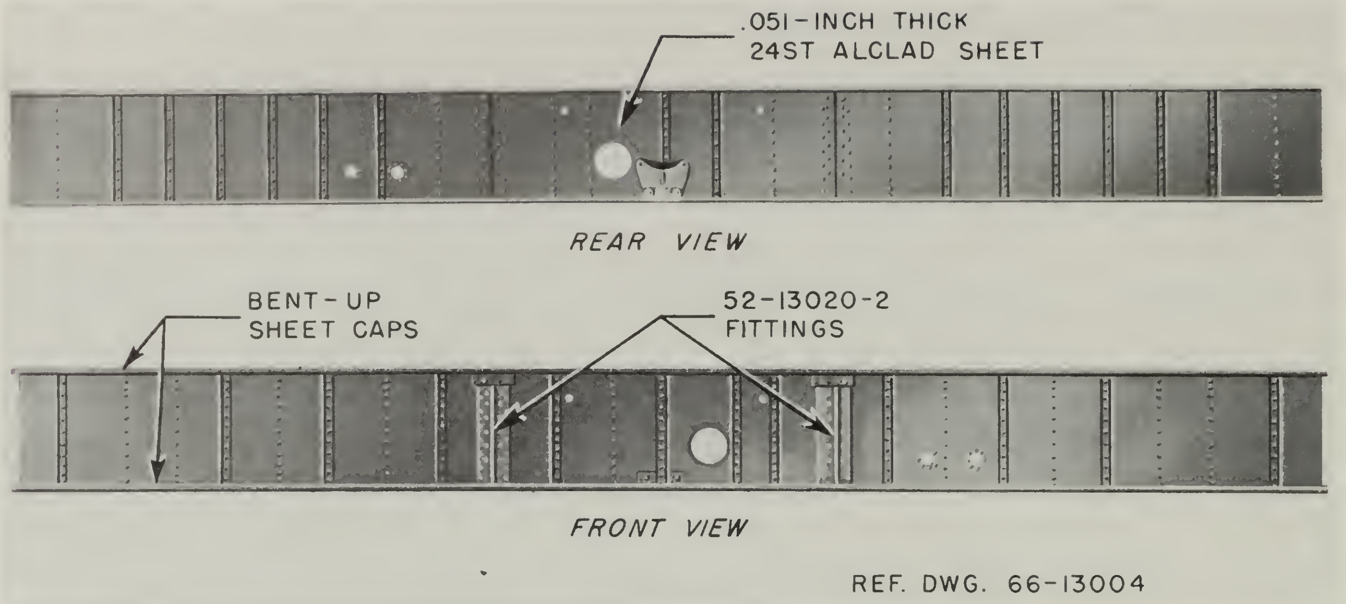


Figure 42—Centersection Rear Spar

spar. In the spar web, center punch the required rivet locations at 3/4-inch spacing. With a No. 21 (.159) drill, drill the center-punched rivet locations and then redrill the existing skin rivet holes through the spar caps. With a No. F (.257) drill, drill the fuel tank door bolt locations through the lower spar cap splice member, using the existing holes in the damaged spar as a guide. Remove the splice members and the spar web stiffeners. Burr all rivet holes. Apply one coat of zinc chromate primer to the overlapping surfaces of the 24ST alclad sheet splice members and to all surfaces of the extruded angle web stiffeners. Reclamp the splice members to the proper positions on the spar. Into the spar web rivet holes, insert and drive AN442-AD5 rivets. Into the skin rivet holes, insert and drive B1227-AD5 rivets. On the lower cap of the rear spar on the AT-6A Series Airplanes, line up B1083 stop nuts with the No. F (.257) holes, and flush rivet the stop nuts to the splice member with AN426-AD3 rivets passed through the existing rivet holes in the skin. On the AT-6C Series Airplanes, instead of the B1083 stop nuts, flush rivet the replacement 22GCH-048 steel channel to the splice member with AN426-AD4 rivets passed through the existing rivet holes in the skin. Secure the fuel tank compartment doors with B1251-428-14 screws to complete the repair *(see Figure 43)*.

33. CENTERSECTION FRONT SPAR UPPER CAP.

In an emergency, damage to the centersection front spar upper C245T cap may be repaired as shown *(see Figure 45)*. It is to be noted that the instructions outlined herein cover damage to the upper extruded spar cap only. If the spar web or the skin over the cap is damaged, refer to the paragraphs containing information pertinent to applicable repairs. This cap repair is an emergency measure to be used only where immediate repair is necessary. Replacement of the cap is recommended. If the cap has been completely severed, cut out the damaged cap material with a hack saw, locating the cut between two skin rivets at each side of the damage. Prepare a filler of Type C245T extrusion to match the removed material. For the splice members, cut two lengths of Type C245T (K16862 Alcoa die) extruded sections. The lengths of the C245T extruded members must be determined by the existing rivet spacing at the affected area multiplied by the number of attachments required at each side of the damage. In most cases, the length of the splice should not exceed 9-1/2 inches plus the length of the damage. Modify the splice members as shown *(see Figure 45)*. With a No. 21 (.159) drill, drill out ten rivets at the side of the damage which secure the cap to the spar web. Take particular care to prevent the elongation of any of the rivet holes. With a No. 30 (.1285) drill, drill out at least nine skin rivets at each side of the damage. Clamp or bolt the C245T angle extrusion splice member to the proper position on the damaged spar cap. With a No. 11 (.191) drill, drill the 38 holes in the splice member, using the existing rivet holes in the cap as a guide. Clamp or bolt the modified C245T splice member plate to the heel of the

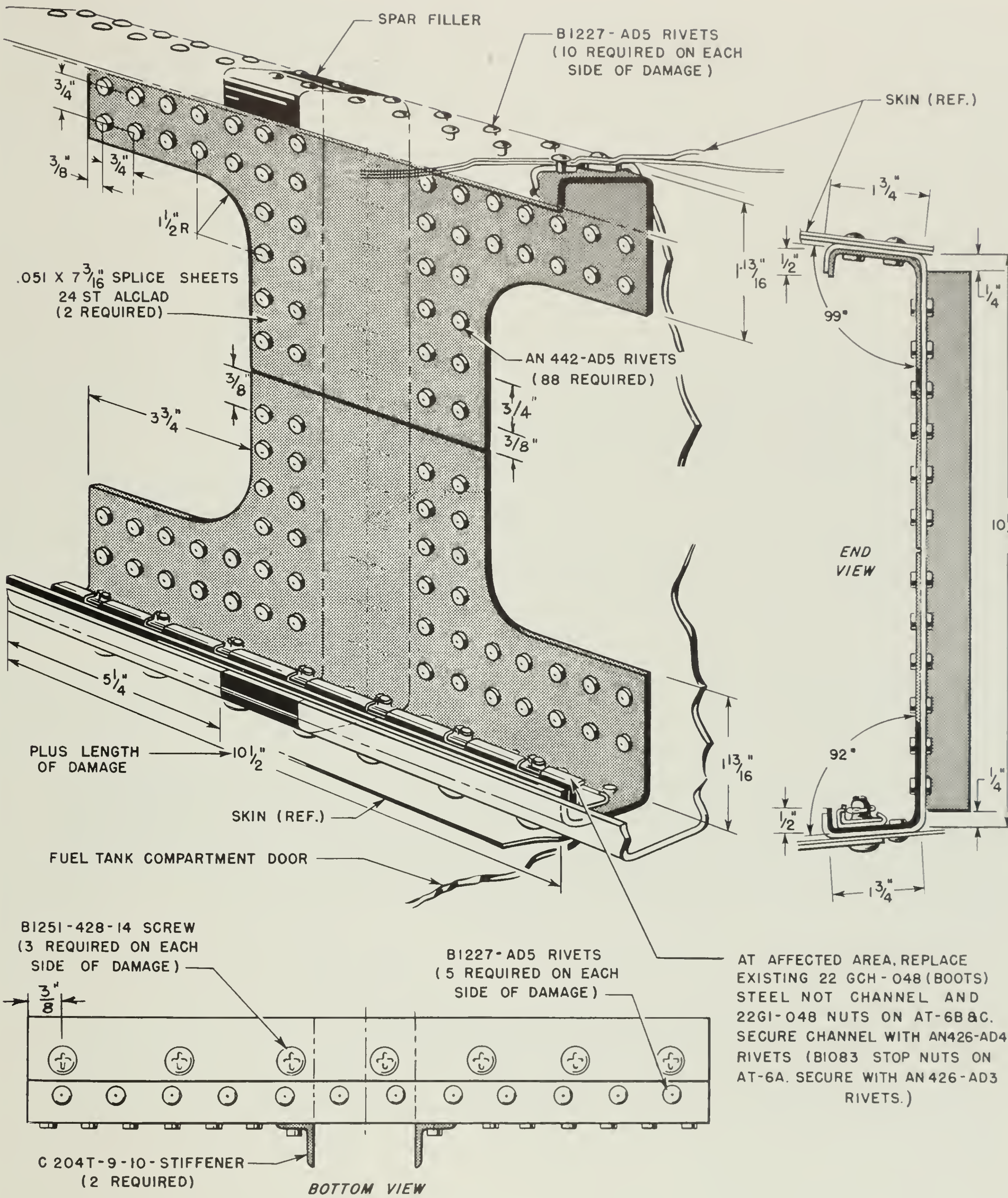


Figure 43—Centersection Rear Spar Splice

cap, and drill twenty No. 11 (.191) holes through the plate, using the existing holes as a guide. All holes at the ends of the splice must have an edge distance of at least 3/8-inch measured from the centerline of the hole to the edge of the material involved. Drill additional holes as required to rivet the filler to the skin. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice members and filler. Again secure the members to the damaged spar cap; and into the eighteen No. 11 (.191) holes through the skin, insert B1251-1032-14 screws and secure with AC365-1032 nuts. Through the twenty No. 11 (.191) holes in the heel of the cap, insert B1251-1032-17 screws and secure with AC365-1032 nuts. In areas where flush rivets were originally employed, substitute B1248-1032 flush screws for B1251 recessed button head screws, after cut countersinking the holes 100 degrees x 3/8-inch diameter. Tighten all nuts with an approximate torque of from 20 to 25 inch-pounds. Bolt threads must not bear in any part of the splice members. Although the screw threads must cut completely through the fiber of the self-locking nut, there should be no more than approximately two screw threads showing beyond the fiber. Normally, no washers should be necessary under the nuts; but if the use of washers is necessary in certain locations where the screw passes through a web stiffener, use AN960-10 washers in the quantity required, and use the next greater length screw. Rivet the C245T spar filler to the splice members with the same size and type rivets as originally employed in the area *(see Figure 45)*.

34. CENTERSECTION FRONT SPAR LOWER CAP.

In an emergency, damage to the centersection front spar lower cap (formed of Type C245T extrusion) may be repaired as shown *(see Figure 46)*. It is to be noted that instructions outlined herein cover damage to the lower extruded spar cap only. If the spar web is damaged, reference should be made to the paragraphs containing information pertinent to an applicable repair. This spar cap repair should be used only in an emergency where immediate repair is necessary. Replacement of a severely damaged spar cap is recommended. Remove the 1/4-inch diameter bolts securing the fuel tank compartment doors. If the cap has been completely severed, cut out the damaged cap material with a hack saw, locating the cut between two fuel tank bolt holes at each side of the

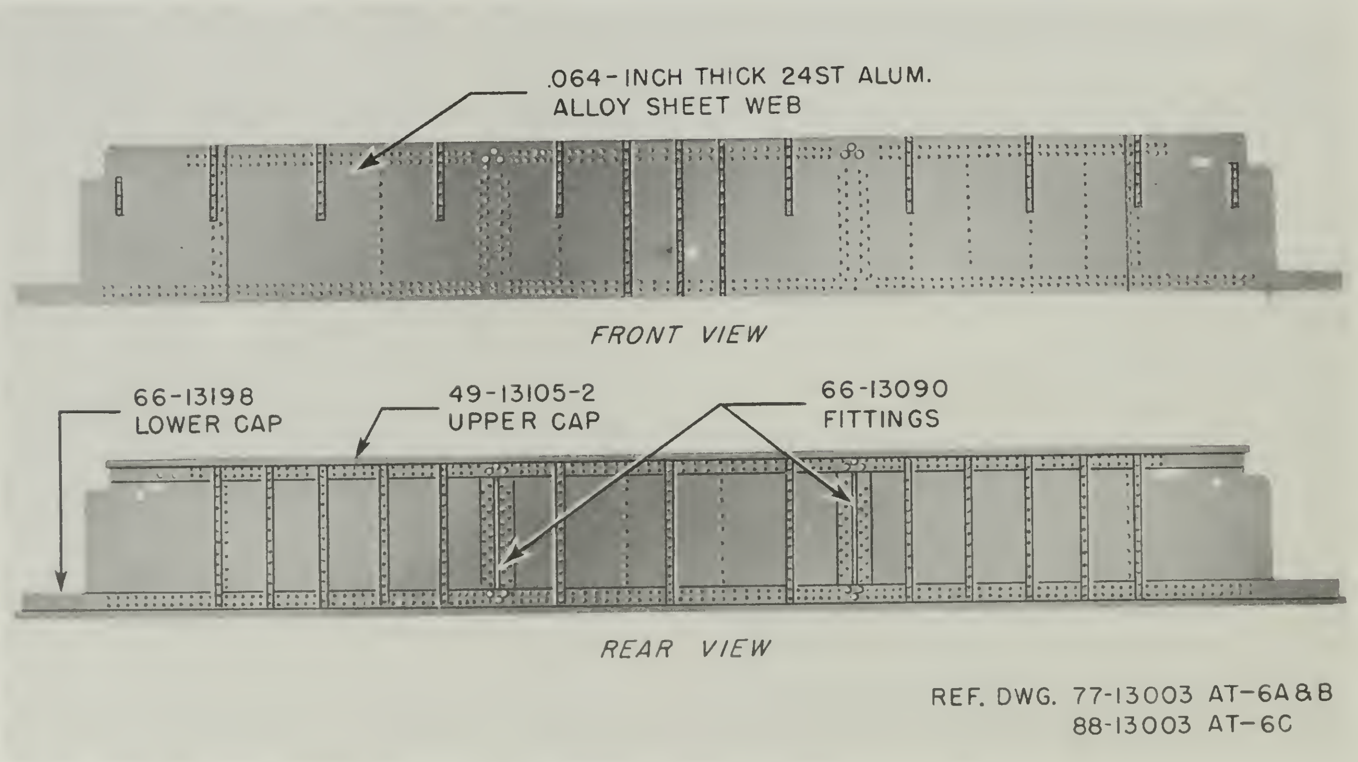


Figure 44- Centersection Front Spar

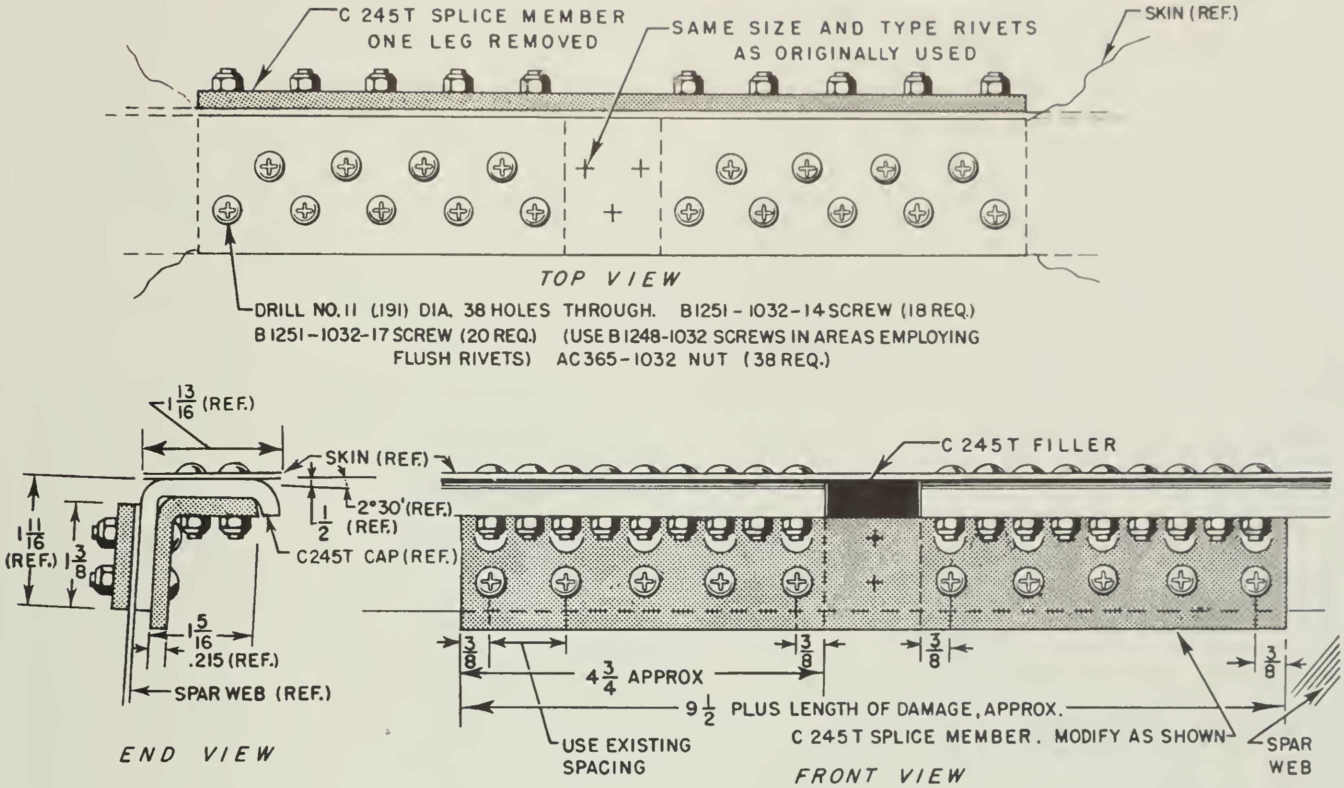


Figure 45—Centersection Front Spar Upper Cap Splice

damage. Prepare a filler section of Type C245T extrusion to match the removed material. For the splice members, cut two lengths of Type C245T (K16862 Alcoa die) extruded sections. The lengths of the C245T extruded members must be determined by the existing rivet spacing at the affected area multiplied by the number of attachments required at each side of the damage. In most cases, the length of the splice should not exceed 13-1/2 inches plus the length of the damage. On one of the extruded splice members, open the existing angle of 92 degrees 30 minutes to an angle of 95 degrees. On the other splice member, cut off one leg to form a 1-3/8 inch wide splice plate. With a No. 21 (.159) drill, drill out the fourteen rivets at each side of the damage which secure the spar cap to the web. Take particular care to prevent the elongation of any of the rivet holes. If any rivets are used through the skin side flange of the spar cap, drill the rivets out with a No. 30 (.1285) drill. On the AT-6A Series Airplanes, drill out the rivets securing the existing B1083 stop nuts in the affected area. On the AT-6B and AT-6C Series Airplanes, the single steel nut channel assembly (NA Dwg. 84-13330) spanning the entire length of the cap must be removed by drilling out the securing rivets. Remove the channel; and with a hack saw, cut out the damaged section of the nut channel 6-3/4 inches from each side of the damage between two of the channel nuts. At each side of the damage, rerivet the two remaining sections of the nut channel to the cap, discarding the damaged section. To replace the damaged section of the nut channel, cut a length of 22GCH-048 steel channel to match the damaged section and snap in the required number of 22G1-048 nuts. (Channel and nuts may be purchased from the Boots Nut Corporation, Waterbury, Conn.) Clamp the C245T angle splice member to the proper position on the spar cap. With a No. F (.257) drill, drill the ten fuel tank compartment door bolt locations, using the existing holes in the cap as a guide. With a No. 11 (.191) drill, drill the 34 holes as shown *(see Figure 46)*, using the existing rivet holes in the cap as a guide. Clamp or bolt the modified C245T splice member plate to the heel of the cap and drill the required holes through. All holes at the ends of the splice must have an edge distance of at least 3/8-inch measured from the centerline of the hole to the edge of the material involved. Drill additional holes as required to secure the filler. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the splice members and filler. Again secure the members and filler to the spar; and into the twenty-eight No. 11 (.191) holes in the heel of the cap, insert B1251-1032-17 screws and secure with AC365-1032 nuts. Into the six No. 11 (.191) holes in the skin side flange of the cap, insert B1251-1032-14 screws and secure with AC365-1032 nuts. Tighten all nuts with an approximate torque of from 20 to 25 inch-pounds. Bolt threads must not bear in any part of the splice members. Although the screw threads must cut completely through the fiber of the self-locking nut, there should be no more than approximately two screw threads showing beyond the fiber. Normally, no washers should be necessary under the nuts; but if the use of washers is necessary in certain locations where the screw passes through a web stiffener, use AN960-10 washers in the quantity required and use the next greater length screw. On the lower cap of the rear spar on the AT-6A Series Airplanes, line up B1083 stop nuts with the No. F (.257) holes and flush rivet the stop nuts to the splice member with AN426-AD3 rivets passed through the existing rivet holes in the skin. On the AT-6C Series Airplanes, instead of the B1083 stop nuts, flush rivet the replacement 22GCH-048 steel channel to the splice member with AN426-AD4 rivets passed through the existing rivet holes in the skin. When installing the fuel tank compartment doors, pass B1251-428-17 screws through the splice members; five of these screws are required at each side of the damage *(see Figure 46)*.

35. PATCHING CENTERSECTION FRONT SPAR WEB.

Damage to the centersection front spar web may be repaired in most cases by patching; or if the damage is severe, the web may be completely spliced as outlined in the following paragraph. Both types of repairs are illustrated *(see Figure 47)*. If the spar web is to be patched, carefully scribe a circle around the damage and with a spiral reamer cut around the scribe line. With a file, smooth out the rough edges of the material. Cut two round sheets of .064 inch thick 24ST alclad that will overlap the spar web hole by 1-1/2 inches all around. Position one of the patches on each side of the spar web. With a No. 11 (.191) drill, drill through the patches and spar web in several locations and temporarily fasten

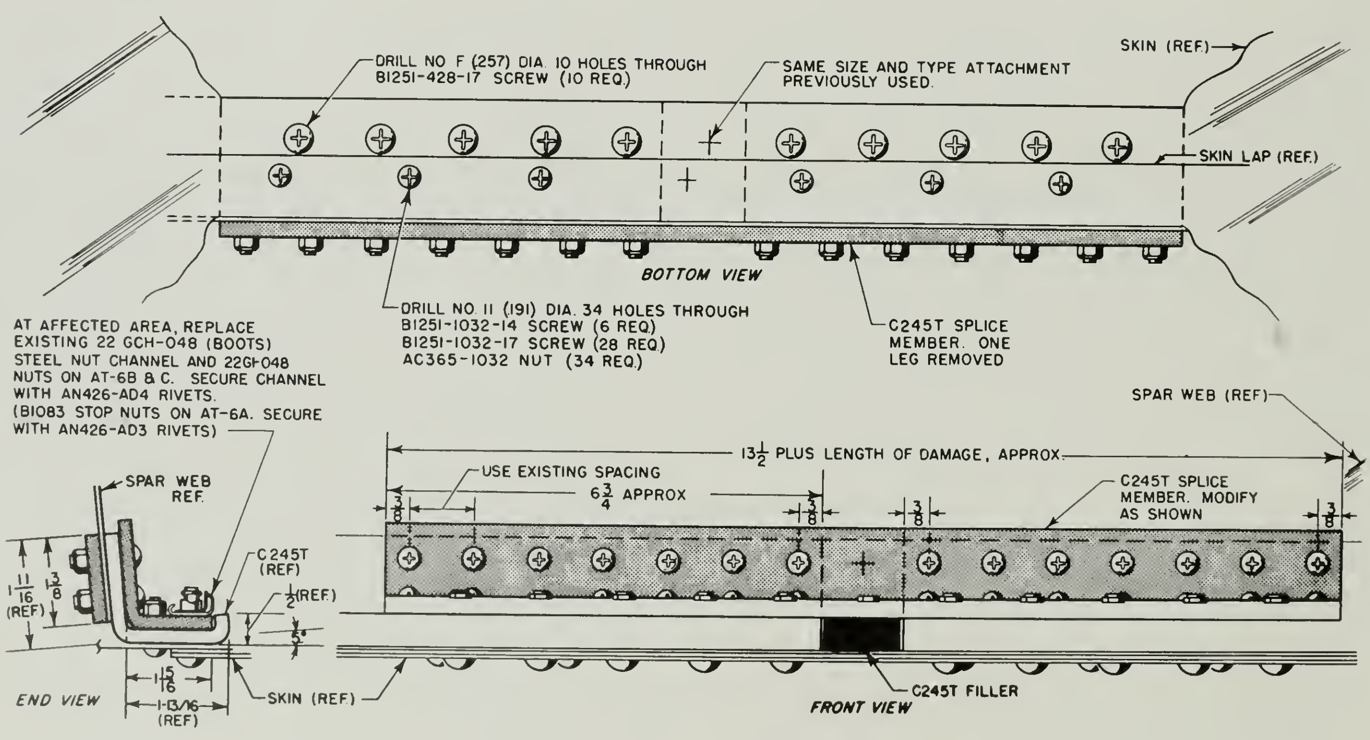


Figure 46—Centersection Front Spar Lower Cap Splice

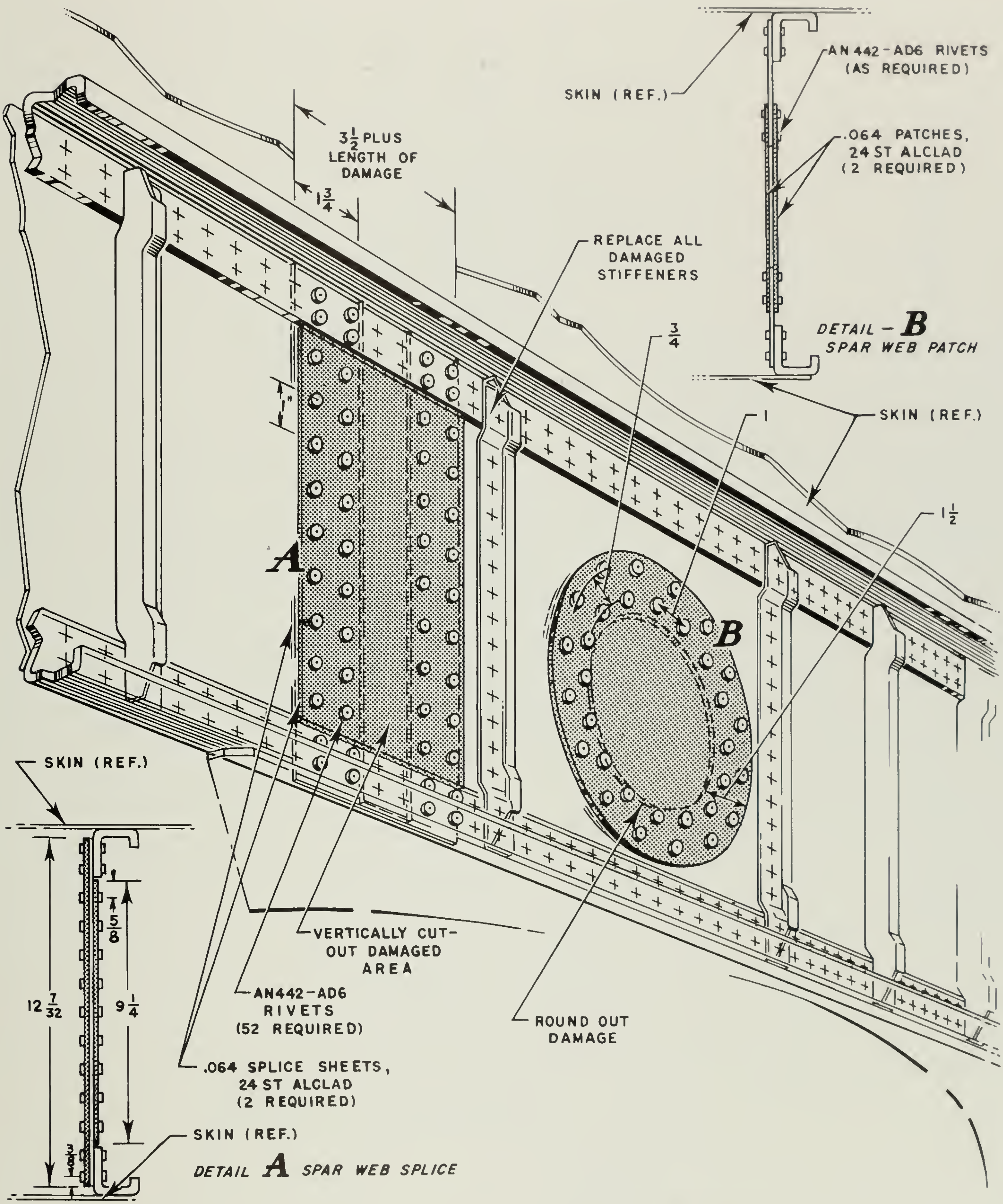


Figure 47—Centersection Front Spar Web Repair

the patches in place with skin fasteners. Drill through the remaining holes in the patches at an average spacing of 1 inch on centers. Remove the patches and burr all the rivet holes. Apply one coat of zinc chromate primer to all surfaces of the patch sheets. Again secure the members to the proper positions on the spar web. Rivet the patch sheets to the spar web in a double shear joint, using AN442-AD6 rivets in the quantity required. If any of the existing web stiffeners in the area are damaged, replace them.

36. SPLICING CENTERSECTION FRONT SPAR WEB.

In cases where patching the spar web will not suffice, the web sheet may be completely spliced as shown *(see Figure 47)*. It is to be noted that the repair outlined herein covers damage to the spar web only. If the extruded spar cap is damaged, reference should be made to the paragraph containing applicable repair information. With a hack saw, make two vertical cuts on each side of the damage, and remove and discard the damaged material. For the splice members, cut two sheets of .064 inch thick 24ST alclad. One of the splice sheets should have a width of 3-1/2 inches plus the width of the damage, and a length of 9-1/4 inches. The other splice sheet should have a width of 3-1/2 inches plus the width of the damage, and a length of 12-7/32 inches. With a No. 21 (.159) drill, drill out the affected rivets in the area, and position one of the patches on each side of the damage. Drill the splice sheets through in several locations and temporarily fasten the sheets in place with skin fasteners. Center punch the locations for two rivet rows at each side of the damage at an average spacing of 1 inch on centers. With a No. 11 (.191) drill, drill the center-punched locations. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice members. Again fasten the members to the spar web. Rivet the splice members in a double shear joint to the spar web with two rows of AN442-AD6 rivets (52 required) at each side of the damage. If any of the spar web stiffeners in the area are damaged, remove them and replace with new stiffeners.

37. VERTICAL STABILIZER FRONT SPAR.

Damage to the front spar of the vertical stabilizer may be repaired as shown *(see Figure 48)*. Depending upon the extent of the damage, all or part of the complete splice illustrated may be used. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. Use a sprial reamer or a hack saw. For the splice members, cut two sheets of .032 inch thick 24ST alclad, each having a length equal to the length of the damage plus 6 inches, and a width of 3-1/4 inches. Along the length of each of the splice members, bend up a 3/4-inch flange, observing a minimum bend radius of 1/8-inch. Trim the splice members to the required shape; and if any flanged lightening holes are present in the affected area, trim the splice members to match the existing lightening holes if necessary. For the web stiffeners, bend up two .032 inch thick 24ST alclad strips equal to the spar depth in length and 1-3/8 inches wide. Drill out the affected skin rivets at each side of the damage. Clamp the splice members to the proper positions on the spar, and drill through and temporarily fasten the splice members with skin fasteners. At each side of the damage, center punch the required rivet locations at an average spacing of 3/4-inch on centers. It is to be noted that all rivets at the ends of the splice must have an edge distance of at least 3/8-inch measured from the centerline of the rivet hole. With a No. 30 (.1285) drill, drill center-punched rivet locations at each side of the damage. With a No. 40 (.098) drill, drill the splice members through the skin holes. Remove the splice members and stiffeners, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice members and temporarily secure the members to the spar. Rivet the splice members to the spar, using AN442-AD4 rivets through the web and replacing the drilled-out skin rivets with AN426-AD3 rivets *(see Figure 48)*. To prevent the splice members from creeping, rivet first at the corners and then at the intermediate positions.

38. VERTICAL STABILIZER REAR SPAR AREA ABOVE ROOT RIB.

Damage to the rear spar of the vertical stabilizer on areas located above the root rib may be repaired as shown *(see Figure 49)*. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice members, prepare two sheets of .040 inch thick 24ST alclad having a length of 6 inches plus the length of the damage and having a maximum width of 6 inches. Along the length of the splice sheets, bend a flange equal to the

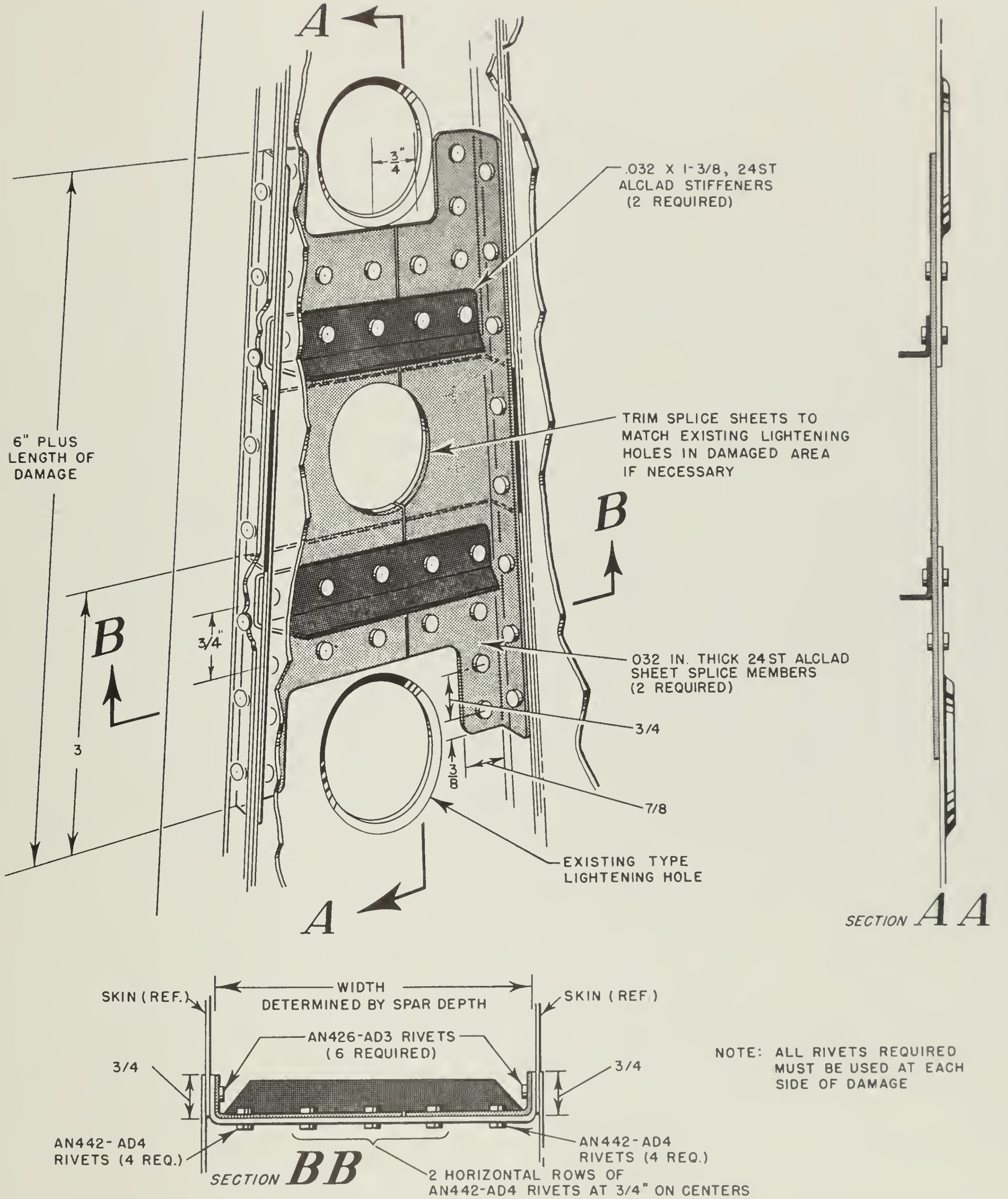


Figure 48—Vertical Stabilizer Front Spar Splice

tapered spar flange; then bend a 90-degree, 29/32-inch flange measured from the first bend. Observe a minimum bend radius of 5/32-inch. Trim the splice members as shown *(see Figure 49)*. For the spar web stiffening angles, bend up two 1-3/8 inch wide strips of .040-inch 24ST alclad of a length equal to the spar depth at the damaged area. With a No. 40 (.098) drill, drill out the affected skin rivets at each side of the damage, taking particular care to avoid elongating the rivet holes. Secure the splice sheets and the web stiffeners to the spar. Drill through the splice sheets in several locations and temporarily secure the sheets to the spar with skin fasteners. Center punch the required sixty rivet locations at an average spacing of 3/4-inch on centers. At the ends of the splice sheets, observe a minimum edge distance of 3/8-inch measured from the center-punch marks. With a No. 21 (.159) drill, drill the required rivet holes. As the spar depth decreases toward the spar tip, omit the rivets on the spar web centerline. With a No. 30 (.1285) drill, drill the splice members through the existing rivet holes in the skin. Remove the splice members and the web stiffeners, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice members. Secure the splice members and web stiffeners to the spar and temporarily secure the members with skin fasteners. Into the sixty No.21 (.159) holes, insert and drive AN442-AD5 rivets. Toward the spar tip, the rivets on the spar centerline and cap flanges may be omitted. Replace the skin rivets with B1227-AD4 rivets in the quantity required *(see Figure 49)*. Rivet first at the corners and then at the intermediate positions to prevent the sheets from creeping.

39. VERTICAL STABILIZER REAR SPAR AREA BELOW ROOT RIB.

The complete splice of the vertical stabilizer rear spar for the area located below the root rib is illustrated *(see Figure 50)*. Depending upon the extent of the damage, all or part of the complete splice may be used. Remove the stabilizer from the fuselage. If the inner reinforcing angles on the spar caps are damaged, drill out the fifty-six AN442-AD4 rivets securing these angles to the spar caps. Use a No. 40 (.098) drill. Remove these angles and cut out the damaged material, locating the cut between two rivet holes at each side of the damage. For the splice members, bend up two sheets of .040 inch thick 24ST alclad having a length equal to the length of the damage plus 7-1/4 inches and having a maximum width of 6-1/2 inches. Along the length of each of the splice members, bend up a flange equal to the tapered spar cap flange at the damaged area; then bend a 29/32 inch, 90-degree flange measured from the first bend. Observe a minimum bend radius of 5/32-inch. Trim the splice members to the shapes shown *(see Figure 50)*. If the two inner angles are damaged, replace the angles with new angles, each bent up from .040 x 3-5/16 x 20-1/4 sheets of 24ST alclad material. Clamp the reinforcing angles, the splice members, and the spar fillers to the spar. At each side of the damage, center punch the required rivet locations at an average spacing of 13/16-inch on centers. Except as noted, observe a minimum edge distance of 3/8-inch measured from the center-punch marks. With a No. 21 (.159) drill, drill the center-punched rivet locations through the splice members. With a No. 30 (.1285) drill, drill the existing rivet holes at each side of the splice members through the inner reinforcing angles. Drill additional rivet holes as necessary. Remove the splice members, filler, and inner reinforcing angle, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice members, filler, and inner angles. Secure the members to the spar and insert a number of skin fasteners in the rivet holes along the length of the inner angle and in various locations in the splice members. At each side of the splice member, replace the drilled-out rivets through the inner reinforcing angles with AN442AD-4 rivets. Through the splice members, insert and drive eighty-eight AN442-AD5 rivets in the pattern noted *(see Figure 50)*. To prevent the splice members from creeping, rivet first in the corners and then in the intermediate positions.

40. REPAIR OF WOODEN HORIZONTAL STABILIZER.

The inaccessibility of the stabilizer prohibits all types of repairs except skin repairs. Refer to Section II for skin repairs to wooden structures. Inasmuch as all members of the stabilizer are glued together, replacement of parts is impractical. Damaged fittings may be replaced *(see Figure 5)*. The following paragraphs on horizontal stabilizer spar repairs are applicable to the aluminum structure only.

41. HORIZONTAL STABILIZER FRONT SPAR.

The front spar of the aluminum horizontal stabilizer is formed of .032 inch thick 24ST

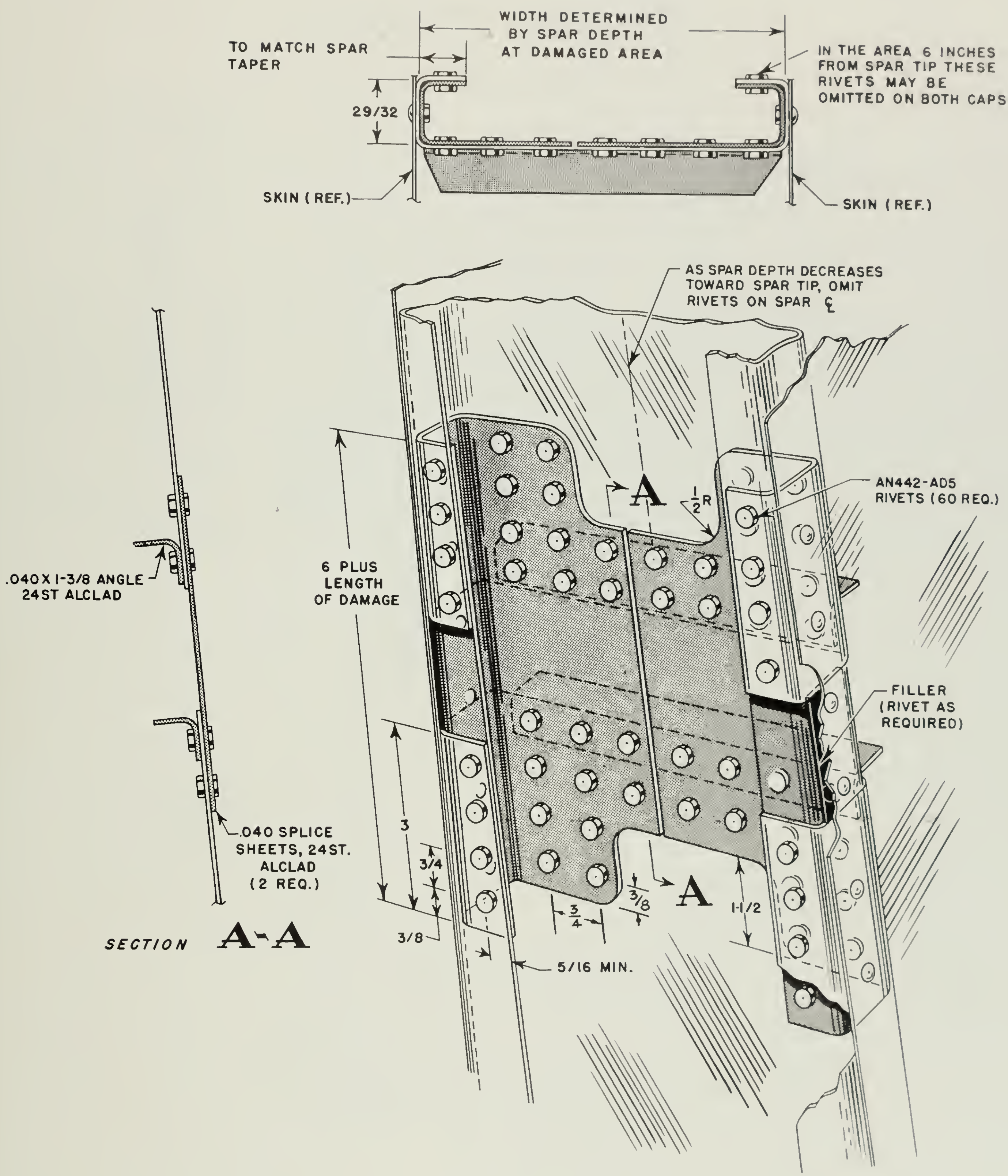


Figure 49—Vertical Stabilizer Rear Spar Splice for Area Above Root Rib

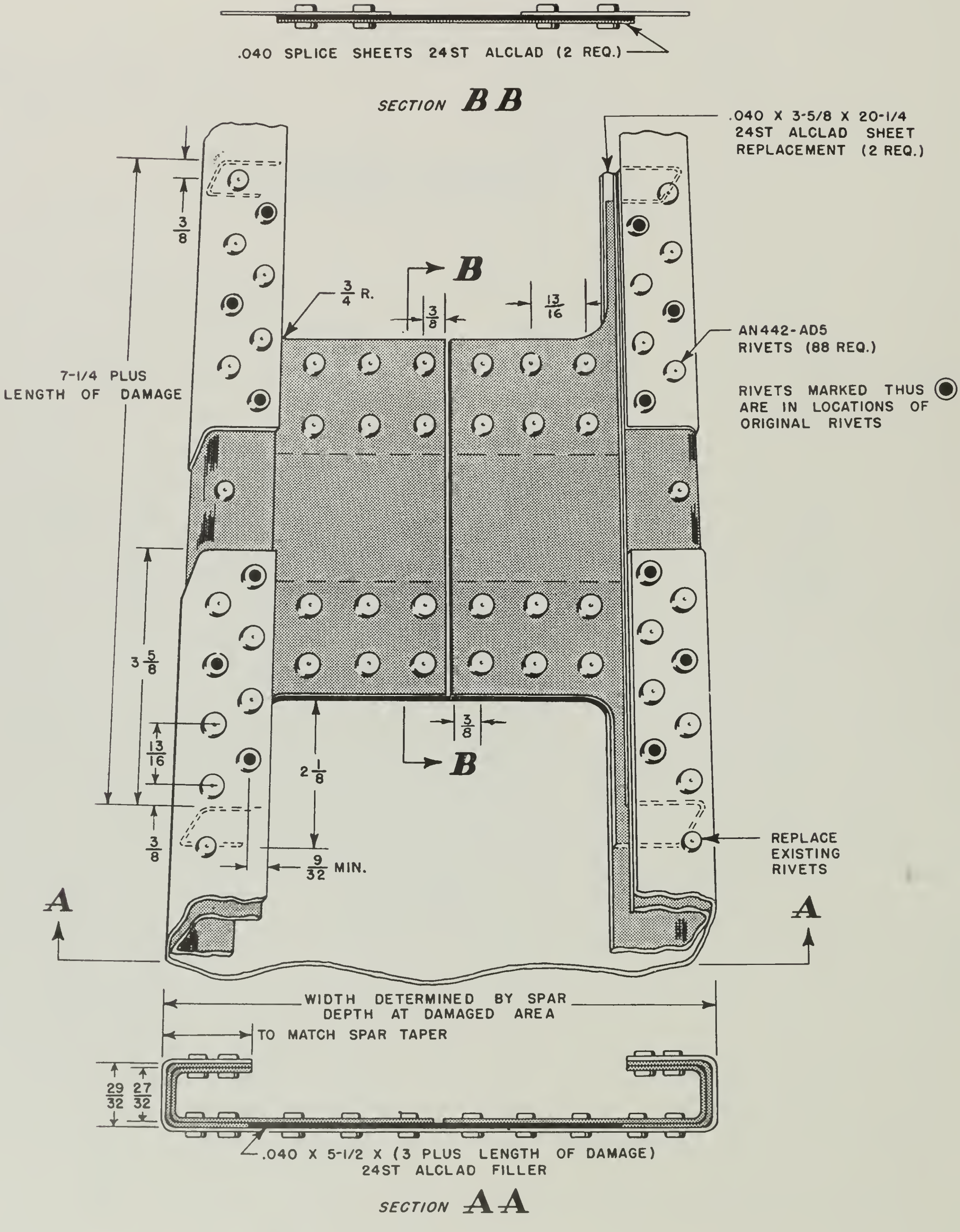


Figure 50—Vertical Stabilizer Rear Spar Splice for Area Below Root Rib

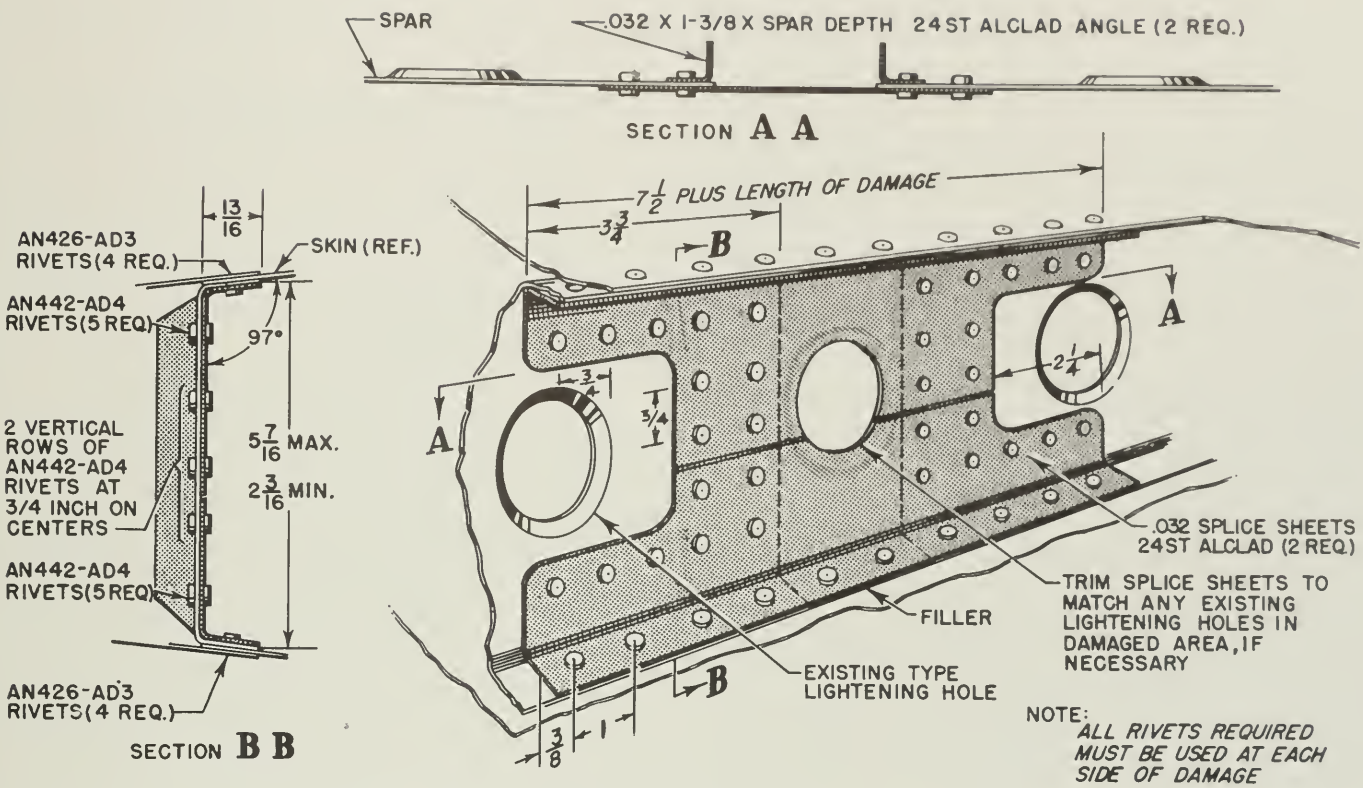


Figure 51—Horizontal Stabilizer Front Spar Splice

alclad sheet. The splice of the member is illustrated *(see Figure 51)*. If the damage is extensive, cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice members, prepare two sheets of .032 inch thick 24ST alclad having a length of 7-1/2 inches plus the length of the damage and having a maximum width of 3 inches. Along the length of the splice sheets, bend a 13/16-inch, 97-degree flange. Trim the splice members to the proper shape. If the damage occurs in areas where flanged cutouts are located, trim the splice sheets to match the holes in the area if necessary. For the spar web stiffener angles, bend up two 1-3/8 inch wide strips of .032-inch 24ST alclad of a length equal to the spar depth. Drill out the affected skin rivets at each side of the damage. Clamp the splice members to the spar. Drill through in several locations and temporarily secure the splice members to the spar with skin fasteners. At each side of the damage, center punch the required rivet locations at an average spacing of 3/4-inch on centers. With a No. 30 (.1285) drill, drill the center-punched rivet locations. At the edges of the sheets, observe a minimum rivet edge distance of 3/8-inch measured from the centerline of the rivet hole to the edge of the material involved. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice members and the web stiffeners. Again secure the members to the spar, and rivet the splice members with AN442-AD4 rivets in the quantity required. Replace the skin rivets with AN426-AD3 rivets in the quantity required *(see Figure 51)*.

42. HORIZONTAL STABILIZER REAR SPAR AREA OUTBOARD OF CENTER HINGE FITTING.

The rear spar of the aluminum horizontal stabilizer in the area outboard of the center hinge fitting may be spliced as shown *(see Figure 52)*. Depending upon the extent of the damage, all or part of the complete splice may be used. Carefully cut out the damaged material, locating the cut between two skin rivets at each side of the damage. For the splice members, cut two sheets of .051 inch thick 24ST alclad, each having a length of 7-1/2 inches plus the length of the damage, and a maximum width of 5 inches. Along the length of each of the splice members, bend up a 90-degree flange equal to the spar flange at the damaged area. Measure 29/32-inch from the bend and then bend another 90-degree flange. Observe a minimum bend radius of 3/16-inch. In areas having flanged cutouts, trim the splice sheets around the holes as required. For the spar web stiffeners, bend up two sheets of .040 inch thick 24ST alclad having a width of 1-3/8 inches and having a length equal to the spar depth at the damaged area. Drill out the affected skin rivets at each side of the damage. Secure the splice members to the spar. Drill through in several locations and temporarily secure the members to the spar with skin fasteners. At each side of the damage, center punch the required rivet locations at an average spacing of 3/4-inch on centers. Observe a minimum edge distance of 3/8-inch measured from the center-punch mark to the edge of the material involved. With a No. 30 (.1285) drill, drill the center-punched rivet locations. Remove the splice members and apply one coat of zinc chromate primer to all overlapping surfaces of the members. Burr the rivet holes. Secure the members to the spar with skin fasteners. Rivet the splice members to the spar with AN442-AD5 rivets in the quantity required *(see Figure 52)*. Rivet first at the corners and then at the intermediate positions.

43. HORIZONTAL STABILIZER REAR SPAR AREA INBOARD OF CENTER HINGE FITTING.

If serious damage occurs to the spar area inboard of the center hinge fitting of the aluminum horizontal stabilizer rear spar, it is advisable to replace the entire spar.

44. SKIN - GENERAL.

Except for the 52S-1/2H aluminum alloy tip skin on the aluminum horizontal stabilizer, all of the skin covering the fixed surfaces consists of 24ST aluminum alloy. The wing skin varies from .020 to .064 inch thick 24ST alclad *(see Figure 53)*. The skin on the stabilizers varies from .020 to .040 inch thick 24ST alclad *(see Figure 54)*. Depending upon the extent of damage, skin panels may be patched, spliced, or replaced as outlined in the following paragraphs. In general, on the flush riveted leading edges, make flush repairs. On the brazier head riveted skin, a flush repair may be desired for appearance, but external

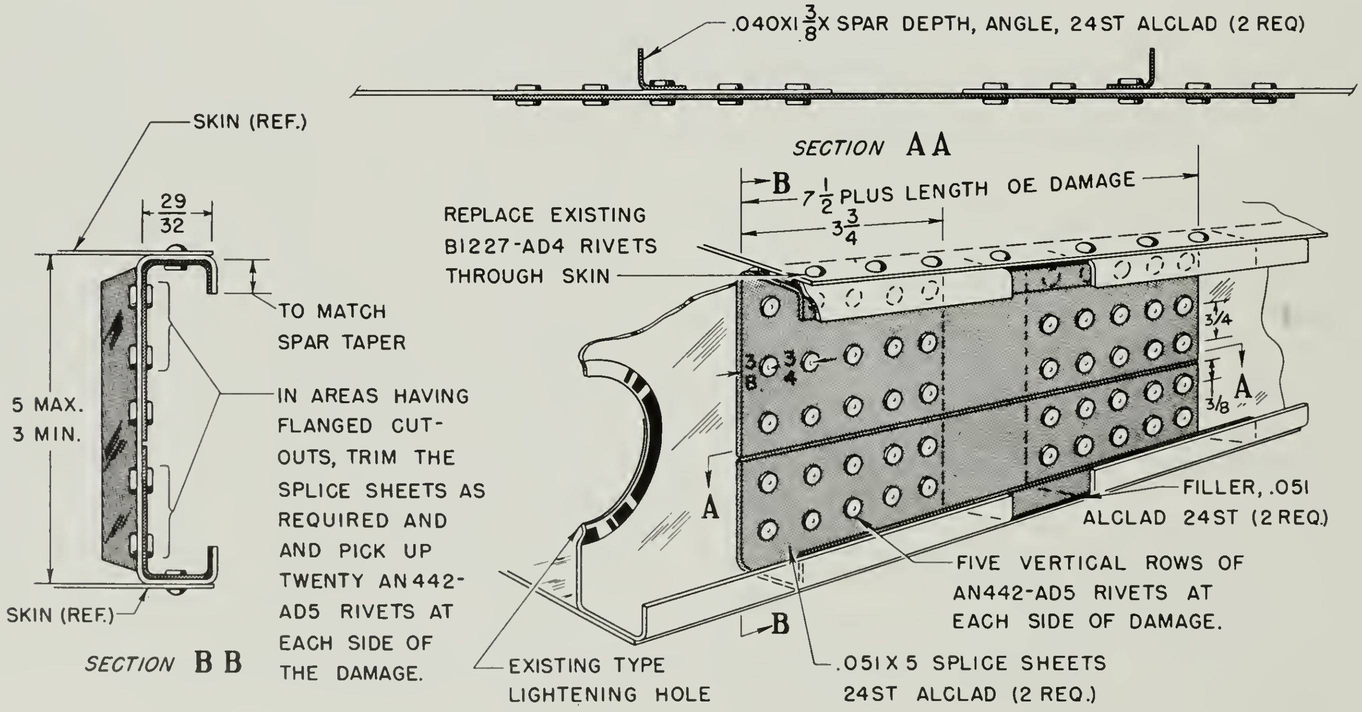


Figure 52—Horizontal Stabilizer Rear Spar Splice for Area Outboard of Center Hinge Fitting

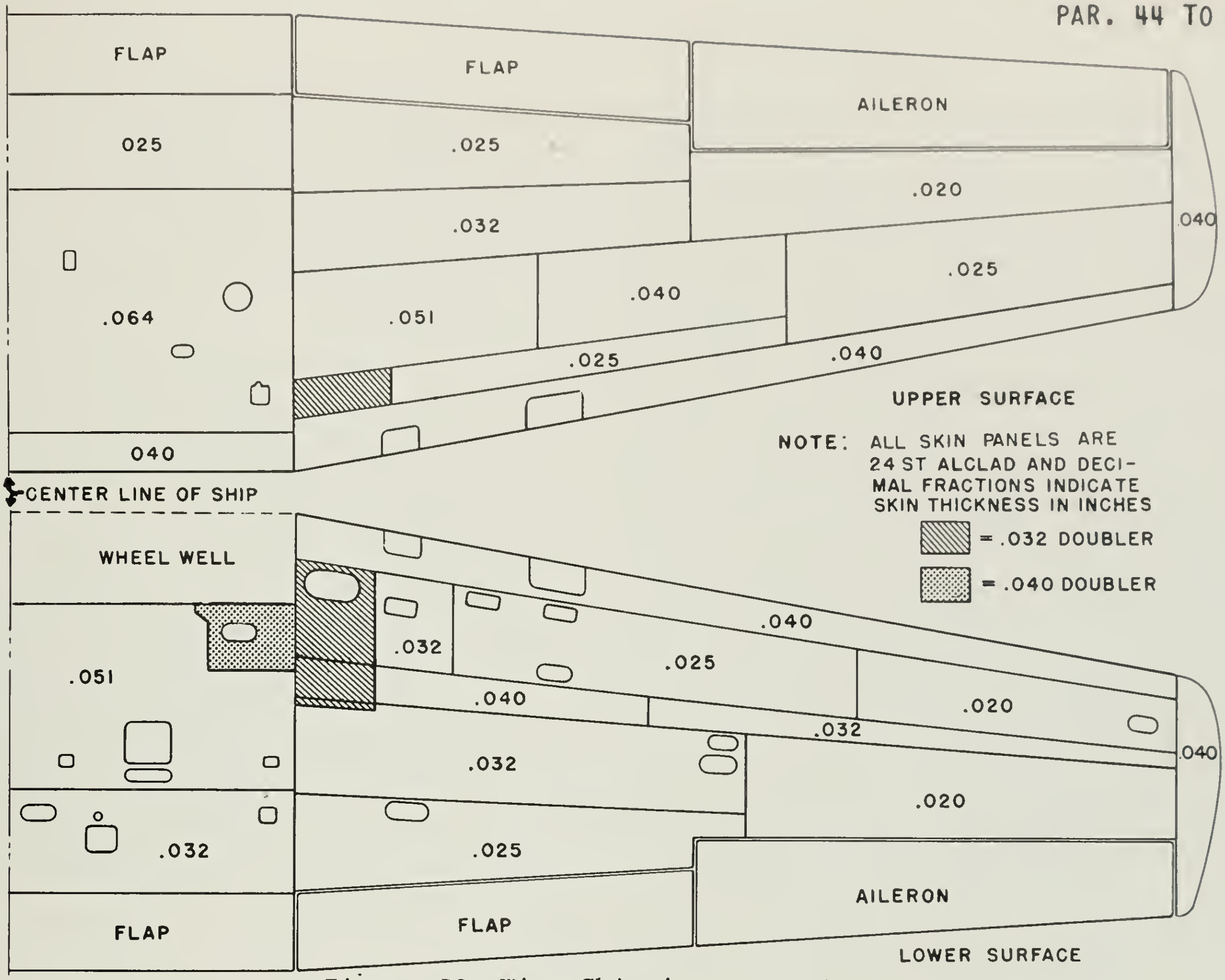


Figure 53—Wing Skin Arrangement

repairs are perfectly acceptable. In areas where interior access is limited, use Cherry blind rivets in place of regular rivets, subject to the restrictions set forth in the following paragraphs.

45. SMALL HOLES IN SKIN LESS THAN 1/2-INCH DIAMETER.

Where the hole is located away from interior structure and is less than 3/16-inch in diameter, fill the hole with the shank of a rivet *(see Figure 55, Detail A)*. Where the hole is located away from interior structure and is up to 3/8-inch in diameter, install a small backing sheet of 24ST alclad and fill the hole with the head of an AN442-AD rivet *(see Figure 55, Detail B)*. Where the hole is located directly over a stringer, drill the hole to the size of the smallest rivet head of a flat head rivet up to and including a 3/16-inch diameter rivet, and insert and drive an AN442-AD rivet through the structure *(see Figure 55, Detail C)*. If the rivet spacing is more than 8 times the diameter of the rivet, add a rivet on each side of the damage, midway between the hole and the adjacent rivets. This repair is especially useful where cracks appear in dimpled skin. It is to be noted that these repairs should not be used within 8 inches of spar or existing cutouts or reinforcement, or within 15 inches of a bolting angle.

46. SMALL HOLES IN SKIN 1/2- TO 1-INCH DIAMETER.

With a hole saw, cut out all small holes in the skin to a round shape *(see Figure 56)*. If the largest dimension of the hole does not exceed 1 inch in diameter, the damage usually may be adequately repaired by means of the small circular button patch illustrated *(see Figure 57)*. Cut the patch from 24ST alclad of the same thickness as the damaged skin. Spot-weld or rivet the filler to the patch. Apply the patch to the skin and drill through the skin and patch in the rivet pattern noted. Use a No. 40 (.098), a No. 30 (.1285), and a No. 21 (.159) drill for 3/32-, 1/8-, and 5/32-inch diameter rivets, respectively. Use the same type rivet (countersunk or brazier head) as is used in the area surrounding the damage. If the hole is located near the intersection of a rib and a stringer, extend the patch to pick up the surrounding rivets in the area *(see Figure 58, Details A and B)*. If the hole is directly over a stringer, the damage may be adequately repaired as shown *(see Figure 58, Detail C)*. For the repair of holes near adjacent structure, the rivet data table is entirely applicable *(see Figure 57)*. It is to be noted that all the patches shown are of the flush type. Whenever the patch is to be slipped between the skin and structure, chamfer edges of the backing sheet to provide a more gradual change in contour. Wherever the smooth surface is not required, the backing sheet may be applied to the outside surface of the skin, and the filler may be omitted. These reinforcements should not be used within 8 inches of a spar, existing cutout, or reinforcement, or within 15 inches of a bolting angle.

47. ISOLATED SKIN CRACKS.

When an isolated crack is found in the skin, drill a No. 40 (.098) hole at the ends of the crack to arrest further growth *(see Figure 59)*. Cut a patch of 24ST alclad large enough to overlap the damage by 1-1/4 inches all around the crack. Fasten the patch in place, drill through the skin and patch, add rivet with two staggered rows of rivets as outlined for patching large holes in skin.

48. PREPARING LARGE HOLES IN SKIN FOR PATCHING.

Large holes in the skin may be repaired by patching if the damage affects less than one-third the width of the skin panel and if the damage is not more than 8 inches wide. Holes which exceed these restrictions should require the replacement of all or part of the skin panel. Extensive skin damage occurring near or directly over existing doublers should require skin panel replacement. Flush patches *(see Figure 60)* should be used on the flush riveted leading edge skin, because external patches introduce unfavorable stalling characteristics. However, for all other locations, the external repair *(see Figure 62)* is recommended for field repairs, because it requires less time and skill to apply than the flush patch. The procedure for both the flush and external skin repair is as follows: With a

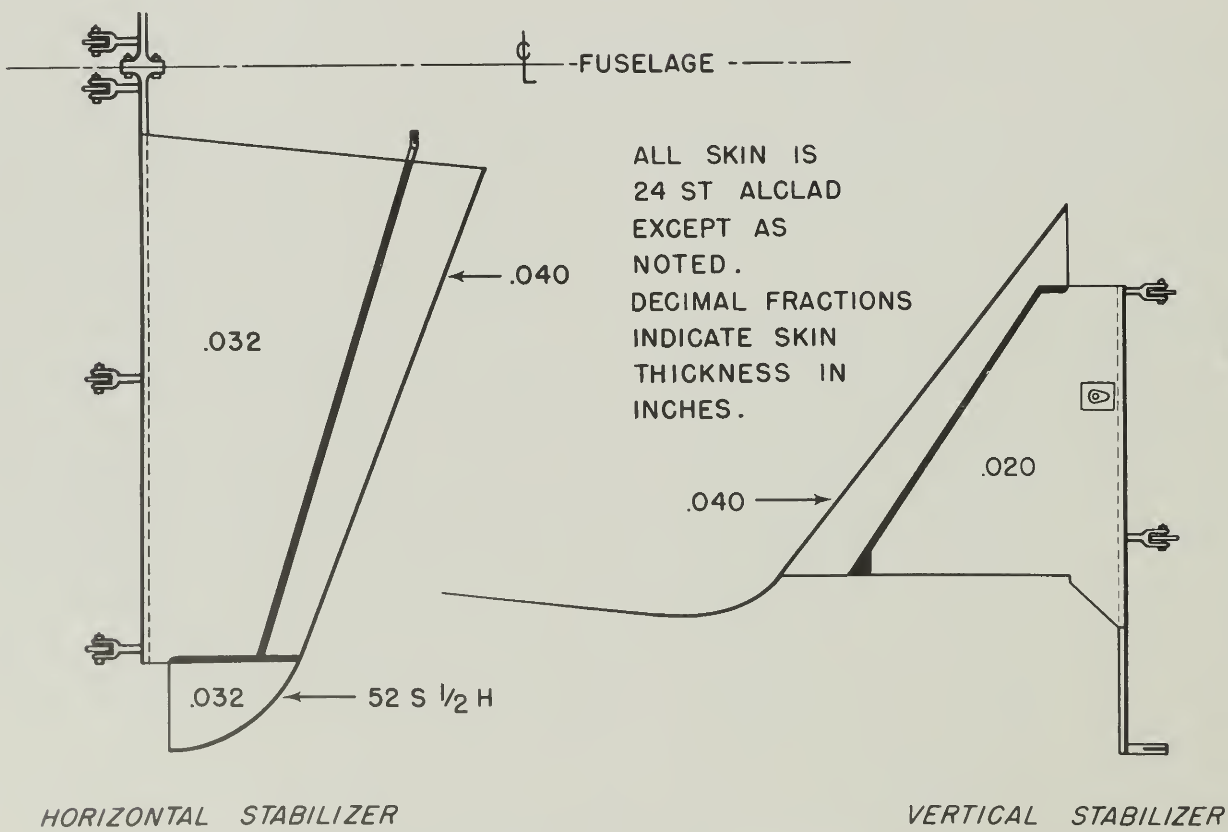


Figure 54—Stabilizer Skin Arrangement

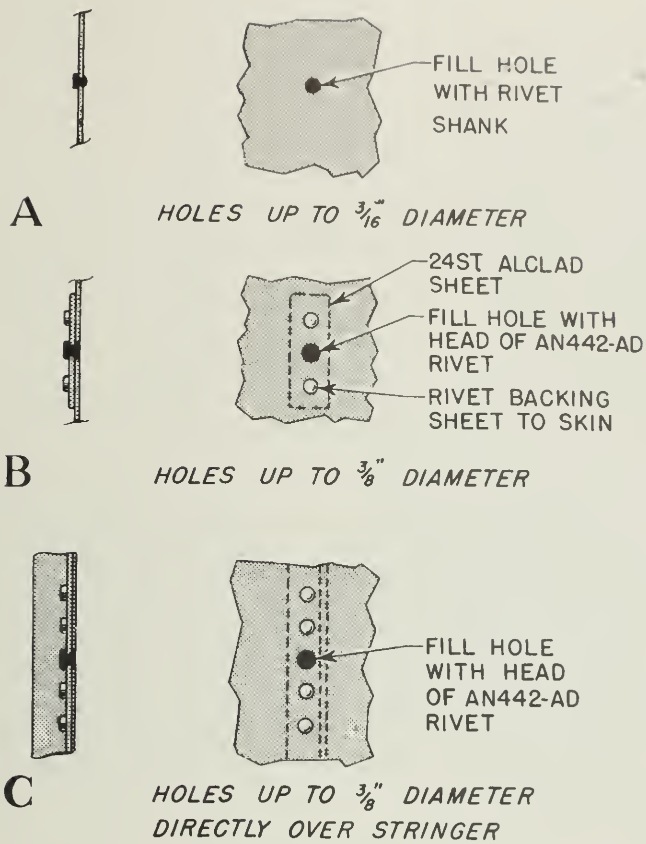


Figure 55—Skin Repair for Holes Less Than 1/2-inch Diameter

grease pencil, mark a rectangle around the damaged area with the sides parallel to the longitudinal stringers. With a pair of dividers, mark each of the four corners with a 1-1/2 inch radius. Cut along the marked line with a spiral reamer or a pair of metal snips, taking particular care to cut smoothly at the radii of all the corners *(see Figure 61)*. Remove the damaged material and smooth out the rough edges of the cut with a file. Obtain a sheet of 24ST alclad of the same thickness as the affected skin panel. Cut the patch to overlap the hole a minimum of 1-1/4 inches all around; but if the hole is located near stringers, the overlap may have to be increased in order to pick up the stringer rivets. Provide all rivet locations around the edges of the patch with an edge distance of 3/8-inch. Chamfer the edges of the patch.

49. PATCHING LARGE HOLES IN SKIN.

After the damage is trimmed and the 24ST alclad patch is cut to the proper size, drill out the existing rivets to be covered by the patch *(see Figure 65)*. Apply the patch to the inside or outside of the skin as required, center punch and drill through the corners of the patch, and temporarily fasten the patch in place with skin fasteners *(see Figure 63)*. If interior access can be gained, drill an outside patch through the existing rivet holes to be covered by the patch. Use the drill from the interior. For an outside patch where it is not possible to manipulate the drill in the interior of the structure, remove the patch and apply a sheet of paper over the area. Fasten the paper by means of masking tape. To locate the rivet holes beneath the paper, blacken the area over each of the holes with a soft pencil, thus outlining the holes on the paper. On the paper, blacken also the locations of the four holes previously drilled to temporarily secure the skin patch *(see Figure 64)*. Next, remove the paper and place the paper on the patch, lining up the four corner holes on the paper and the patch. Center punch the rivet locations on the patch through the paper. Remove the paper, and drill the patch with the size drill required for the thickness of the skin as outlined below. Secure the patch to the skin with skin fasteners and mark a line around the patch 3/8-inch in from the patch edge; then mark a rivet line around the patch 3/8-inch in from the edge of the cutaway damage. It is to be noted that the procedure may have to be varied where the

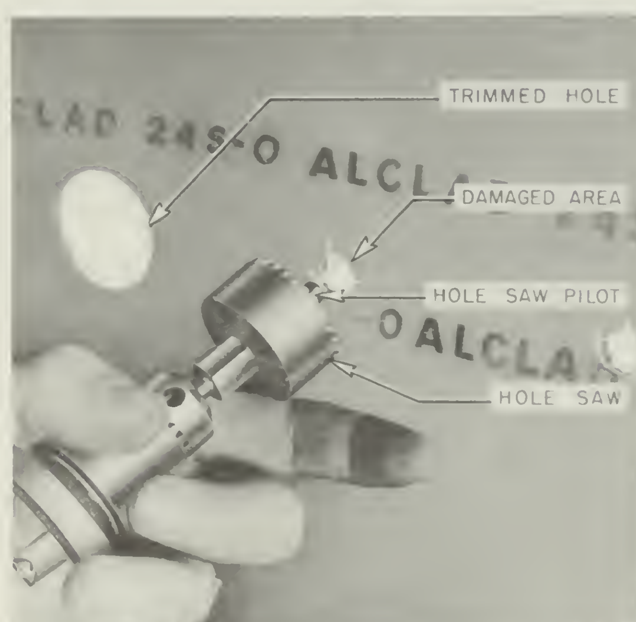
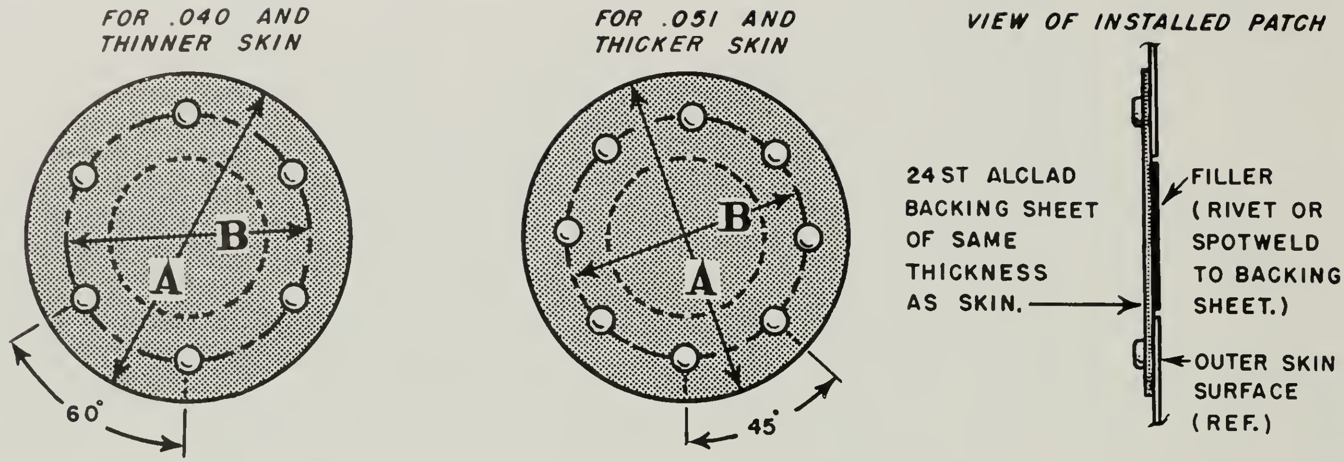


Figure 56—Using Hole Saw To Trim Skin Damage



MAT'L.	1/2 IN. DIA. HOLE				3/4 IN. DIA. HOLE				1 IN. DIA. HOLE			
GA. (IN.)	A (IN.)	B (IN.)	RIVETS SIZE	RIVETS NO.	A (IN.)	B (IN.)	RIVETS SIZE	RIVETS NO.	A (IN.)	B (IN.)	RIVETS SIZE	RIVETS NO.
.020	1-3/4	1-1/8	AD3	6	2	1-3/8	AD3	6	2-1/2	1-3/4	AD3	6
.032	1-3/4	1-1/8	AD4	6	2	1-3/8	AD4	6	2-1/2	1-3/4	AD4	6
.040	1-3/4	1-1/8	AD4	6	2	1-3/8	AD4	6	2-1/2	1-3/4	AD4	6
.051	1-3/4	1-1/8	AD5	8	2	1-3/8	AD5	8	2-1/2	1-3/4	AD5	8
.064	1-3/4	1-1/8	AD5	8	2	1-3/8	AD5	8	2-1/2	1-3/4	AD5	8

Figure 57—Patch for Skin Holes 1/2- to 1-inch Diameter

patch picks up adjacent stringer rivets. Center punch the rivet locations in the approximate pattern noted below for the thickness of skin involved. Using the appropriate drill size noted below, drill the center-punched rivet locations. If stringer rivet holes picked up beyond the damage are spaced farther apart than 1-1/4 inches on centers, drill a hole through the stringer and patch, between each of the existing holes. Remove the patch and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces and secure the patch to the damaged skin with skin fasteners. Rivet the outside patch in place with B1227-AD rivets *(see Figure 62)*. If a flush skin patch is to be applied, slip the patch beneath the skin, and rivet a filler over the damage *(see Figure 60)*. The following chart specifies the rivet size, the rivet rows, rivet spacing, and the drill size to be used about the perimeter of the skin patch. It is to be noted that the rivet rows must be staggered. Use Type B1227-AD brazier head rivets in the quantity required. Where blind riveting is necessary because of limited access, use LS1129 Cherry blind rivets of the same diameter as below and grip length required.

THICKNESS OF SKIN IN INCHES	RIVET TYPE AND SIZE	RIVET PATTERN (ON CENTERS)	SPACE BETWEEN ROWS	DRILL
.020	AD3	2 ROWS AT 5/8-IN.	1/2-IN.	NO.40
.025	AD4	2 ROWS AT 5/8-IN.	1/2-IN.	NO.30
.032	AD4	2 ROWS AT 5/8-IN.	1/2-IN.	NO.30
.040	AD5	2 ROWS AT 5/8-IN.	1/2-IN.	NO.21
.051	AD5	2 ROWS AT 5/8-IN.	1/2-IN.	NO.21
.064	AD6	2 ROWS AT 3/4-IN.	5/8-IN.	NO.11

50. LOCATING BLIND RIVET HOLES IN SKIN.

In some cases where interior access is limited and a skin patch covers existing rivet holes in the skin, a method of locating the existing rivet holes in the skin through the patch is necessary. A makeshift way of locating blind holes, using a sheet of paper as a template, is described in the preceding paragraph. However, in place of the paper method, a permanent blind rivet hole locating tool can be made from scrap material with very little effort *(see Figure 66)*. This tool is quite useful and saves much time in the application of patches where blind hole drilling is necessary. The tool is made up of two strips of thin 24ST alclad, 3/4-inch wide and 6 inches long, riveted together at one end with several AN442-AD4 rivets. At the other end of the strips, a No. 40 (.098) pilot hole is line-drilled through both sheets. In one of the strips, drill through the pilot hole with a drill equal to the size of the blind holes to be located. It is to be noted that it is desirable to make a separate tool for each of the four common rivet diameters encountered; namely, 3/32, 1/8, 5/32, and 3/16. Into the hole in one of the strips of the same size as the blind holes, insert an AN442-AD rivet of the proper diameter. Stake the shank of the rivet to the sheet. In application, slip the patch sheet between the two strips of the tool. Manipulate the tool so that the rivet shank in the tool drops into the rivet hole

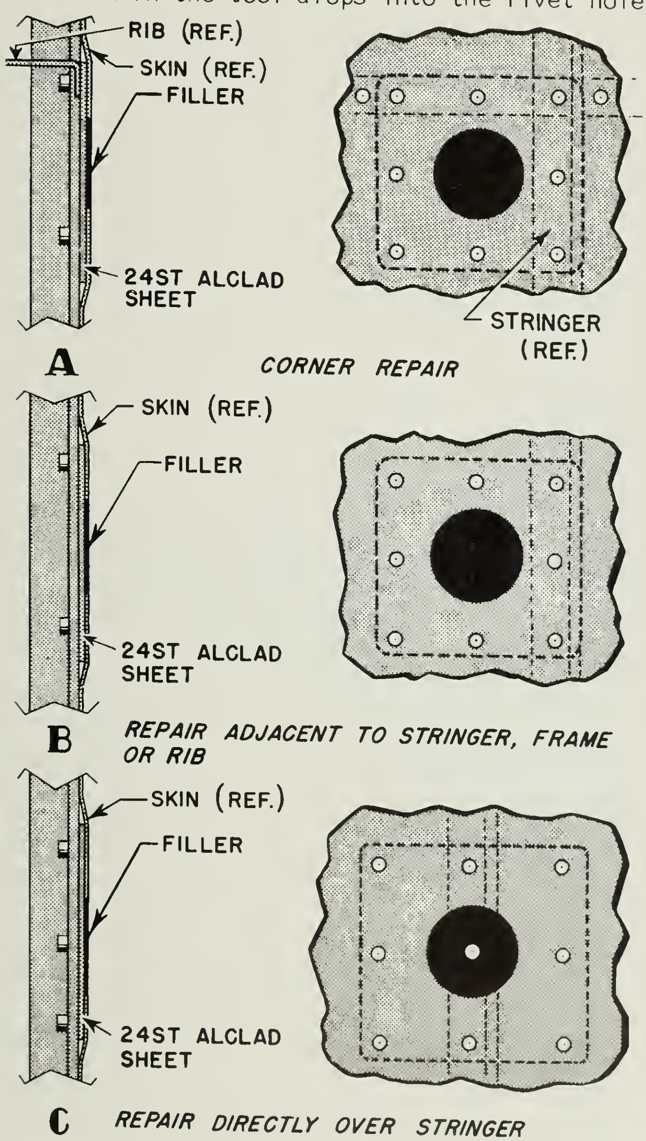


Figure 58—Patch for 1-inch Diameter Skin Holes Near Adjacent Structure

Figure 59—Arresting Growth of Crack

covered by the patch sheet *(see Figure 67)*. When this occurs, mark rivet location on the patch sheet through the guide hole in the top sheet of the tool. Certain types of blind rivet hole locating tools employ a pilot bushing instead of a pilot hole *(see Figure 68)*.

51. SPLICING SKIN PANELS.

Holes in the skin that exceed one-third the width of the skin or exceed 8 inches in diameter should require the complete replacement of the damaged panel. However, if the damage is localized and complete replacement is not warranted, it is sometimes permissible to cut the entire damaged panel and to splice a replacement skin panel into position. It is to be noted that no splice should be made in the same bay with the end of the sheet. Never should two splices be made in the same bay, and it is not permissible to make a spanwise splice within 15 inches of the spars. Either a spanwise or a chordwise splice may be employed; but since practically all the skin panels are considerably shorter in a chordwise direction, a chordwise splice is recommended for general repair, for it entails less work *(see Figure 60)*. Properly support the affected section of the structure, and drill out all the rivets securing the damaged portion of the skin panel to the surrounding structure *(see Figure 65)*. Take particular care to prevent the elongation of

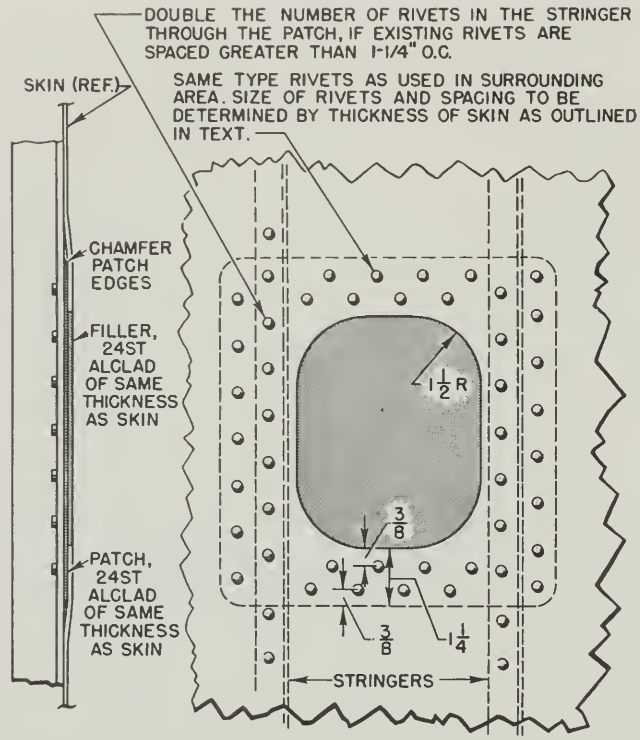


Figure 60—Typical Flush Patch for Large Skin Holes

the existing rivet holes. Pull back the damaged portion of the skin panel; and with a pair of snips, cut the entire damaged skin panel in a chordwise direction. Cut a sheet of 24ST alclad of the same thickness and dimensions as the damaged portion of the skin panel, but extend the dimension of the replacement sheet a minimum of 2 inches beyond the cut edge of the damaged skin panel on .040 inch thick skin. On .032-inch skin and thinner, the splice overlap may be decreased to 1-3/4 inches. Place the new skin section beneath the damaged skin section that was removed. Drill the rivet holes in the new skin section, using the existing rivet holes in the damaged panel as a guide. Secure the new skin panel to the structure and fasten with skin fasteners. Where the replacement skin panel section overlaps the cut edges of the original skin panel, mark a line in from the patch edge 3/8-inch and then mark the two remaining rivet lines 1/2-inch apart on .032 inch thick skin and thinner or 5/8-inch apart on .040 inch thick skin. Drill through the opposite ends of the splice, and fasten the splice overlap in place with skin fasteners. Along the three marked rivet lines of the skin overlap, center punch the required rivet locations as noted below for the thickness of the skin involved. Drill the overlap rivet locations with the size drill noted below. Remove the replacement skin panel and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again secure the skin replacement panel to the structure. Rivet the replacement skin portion to the surrounding structure with the same type and size rivets previously removed from the damaged skin. Rivet the splice overlap with the rivet size, rivet spacing, and rivet rows noted below *(see Figure 69)*. Do not stagger the rivets in the rows.

THICKNESS OF SKIN IN INCHES	RIVET TYPE AND SIZE	RIVET PATTERN (ON CENTERS)	SPACE BETWEEN ROWS	DRILL
.020	AD3	3 ROWS AT 5/8-IN.	1/2-IN.	NO.40
.025	AD4	3 ROWS AT 5/8-IN.	1/2-IN.	NO.30
.032	AD4	3 ROWS AT 5/8-IN.	1/2-IN.	NO.30
.040	AD5	3 ROWS AT 3/4-IN.	5/8-IN.	NO.21
.051*				
.064*				

*Because of the shape and location of these panels on the wing centersection, complete replacement is recommended.

Figure 61—Using Spiral Reamer To Trim Skin Damage

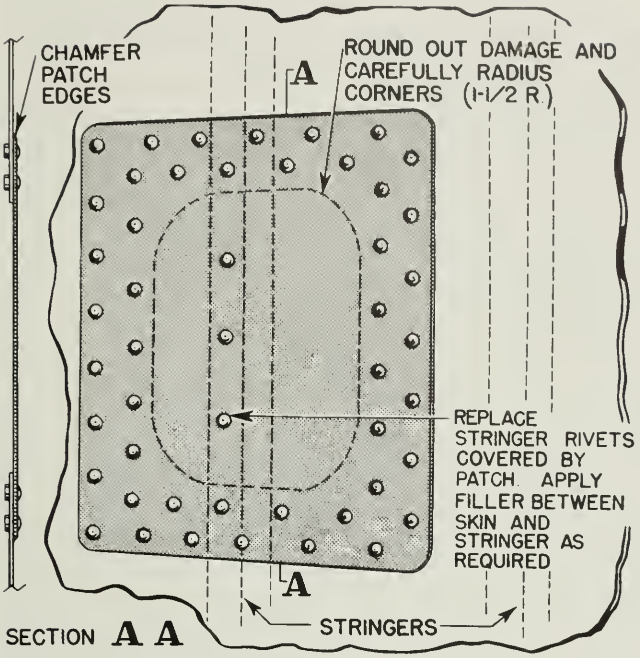


Figure 62—Typical External Patch for Large Skin Holes

52. PREPARING WING LEADING EDGE SKIN FOR SPLICING.

If the skin forward of the leading edge stringer is severely damaged, the damage should be repaired by splicing in an entire leading edge skin replacement sheet as outlined herein *(see Figure 70)*. At each side of the damage, cut into the leading edge skin with a hack saw,

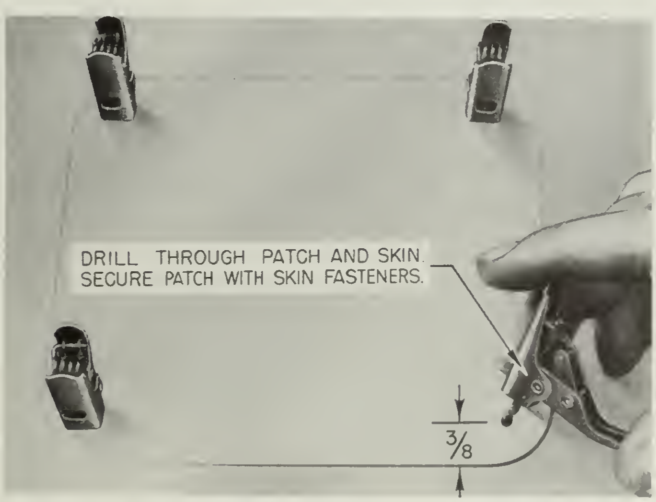


Figure 63—Temporarily Securing Skin Patch Prior to Riveting

Figure 64—Locating Blind Rivet Holes by Improvised Method

locating the cut within 2 inches of the nearest undamaged nose rib. With a No. 40 (.098) drill, drill out all the rivets securing the damaged skin to the structure. Take particular care to avoid elongating any of the existing rivet holes in the skin. Lift the damaged portion of the skin free from the surrounding structure. If any of the nose ribs are damaged, see the applicable paragraph for

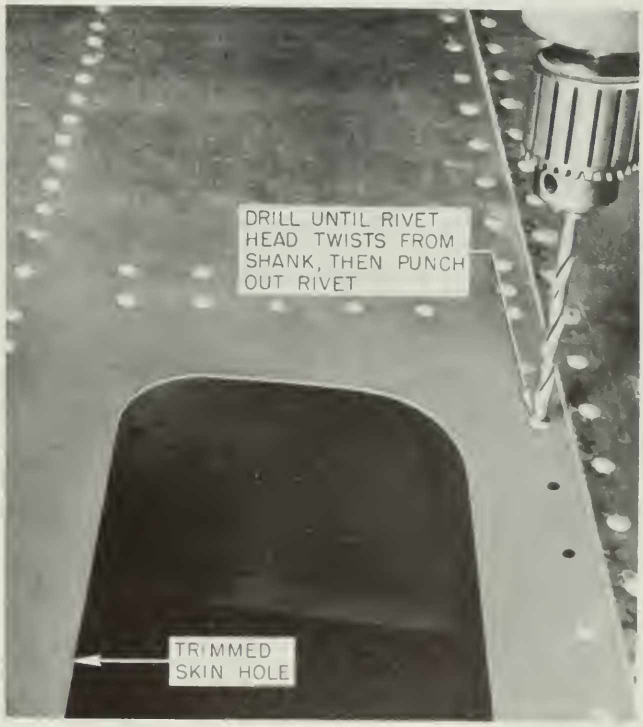


Figure 65—Drilling Out Rivets Covered by Skin Patch

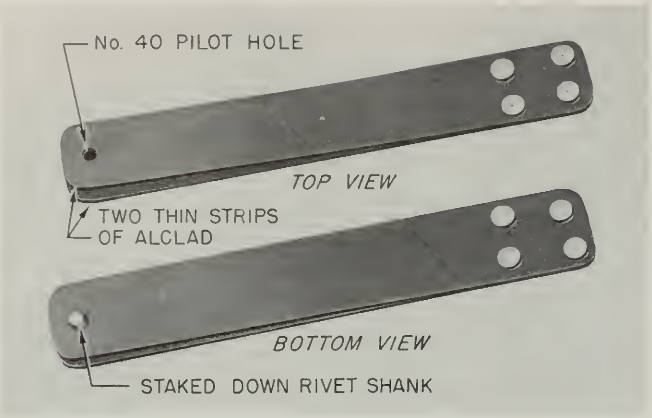


Figure 66--Blind Rivet Hole Locating Tool

rib repairs. For the splice members, cut two sheets of .040 inch thick 24ST alclad, each having a maximum length of 27 inches and a width of 3 inches. Along the length of the splice sheets, bend up 1/4-inch flanges, observing a minimum bend radius of 1/8-inch. In the center six inches of the splice sheets, taper cut the flanges and drill a No. 40 (.098) relief hole at the termination of the flanges. Form the splice sheets to the contour of the leading edge as shown *(see Figure 70)*. At each

Figure 67—Using Blind Rivet Hole Locating Tool

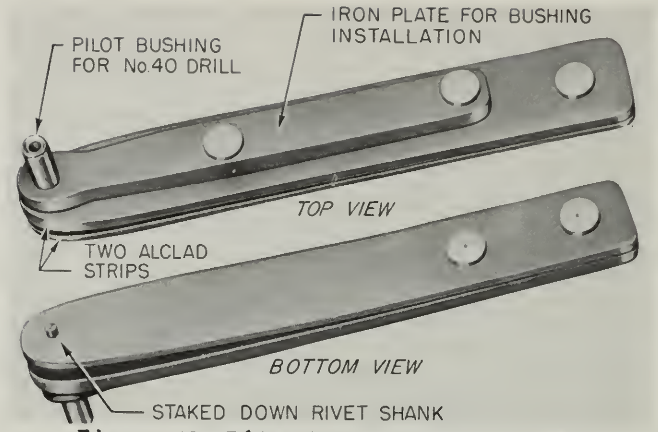


Figure 68—Blind Rivet Hole Locating Tool With Pilot Bushing

of the cuts in the leading edge, clamp the splice sheets, allowing half of the width of the sheets to extend beyond the cut. On each of the splice locations, mark a line 3/8-inch in from the edge of the cut and then mark another line around the leading edge contour 7/8-inch in from the cut. Along each of these lines, center punch rivet locations at an average spacing of 3/4-inch on centers and stagger the rivet locations in the rows. With a No. 30 (.1285) drill, drill the center-punched rivet locations through the skin and the splice sheets. Countersink the rivet holes 100 degrees by 7/32-inch. Into these holes, insert and drive AN426-AD4 rivets in the quantity required. Thus far in the repair, the damaged nose skin is cut away and the splice sheets are riveted to the stubs of the leading edge.

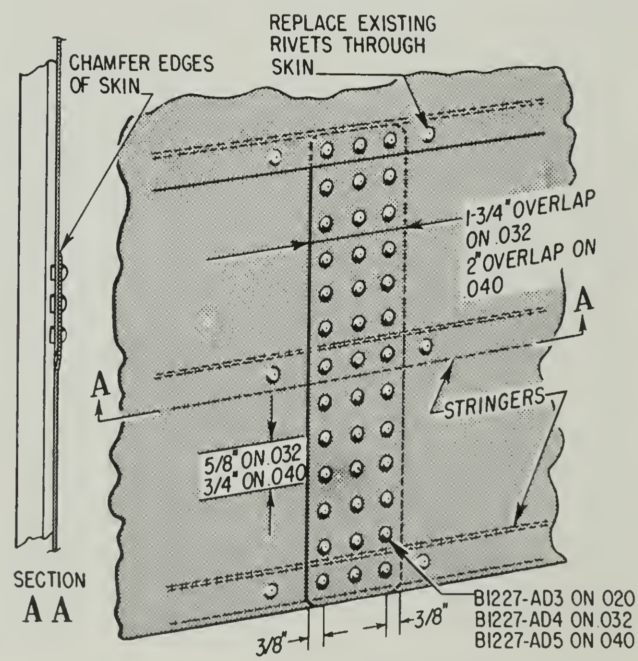


Figure 69--Typical Skin Splice

53. SPLICING WING LEADING EDGE SKIN.

By following the procedure in the preceding paragraph, the damaged leading edge skin is prepared for splicing. Next, for the nose skin replacement panel, cut a sheet of .040 inch thick 24ST alclad having a maximum width of 27 inches and a length equal to the length of the damage. Roll the replacement skin panel to the contour of the leading edge, and trim the sheet to fit the leading edge. If several bays of the leading edge are crushed in, check the contour of the leading edge replacement skin with the correct leading edge contours shown *(see Figure 71)*. Note the station location of the damage *(see Figure 10)*, and then check the exact contour at the station *(see Figure 71)*. Lay out the noted contour in full size upon a panel of 1/2-inch plywood and cut around the contour with a band saw. Use this full-size template to check the contour of the leading edge skin replacement. Position the replacement skin panel in the correct location on the upper side of the leading edge and drill upward through the stringer and sheet with a No. 30 (.1285) drill. If the area is otherwise inaccessible for the drilling, the lower edge of the replacement panel may be pulled away from the structure to provide access. Insert skin fasteners through the upper skin and stringer. Along the lower contour of each end of the skin replacement panel, mark a line 3/8-inch in from the edge of the sheet and then mark a second line along the contour 7/8-inch in from each edge of the replacement panel *(see Figure 70)*. Along each of the lines, lightly center punch rivet locations at an average spacing of 3/4-inch on centers, and stagger the rivet locations in each row. With a No. 30 (.1285) drill, drill the center-punched rivet locations, and insert skin fasteners. With a No. 30 (.1285) drill, drill downward through the lower stringer and skin, using the existing rivet holes in the stringer as a guide. If it is necessary to gain access to the lower stringer for drilling, remove the skin fasteners securing the upper edge of the skin to the upper stringer and pull back the upper edge of the sheet. Along the upper contour of each end of the skin replacement panel, mark a line 3/8-inch in from the edge of the sheet and then mark a second line along the contour 7/8-inch in from each edge of the replacement panel. Center punch the rivet locations and drill No. 30 (.1285) holes in each row at an average spacing of 3/4-inch on

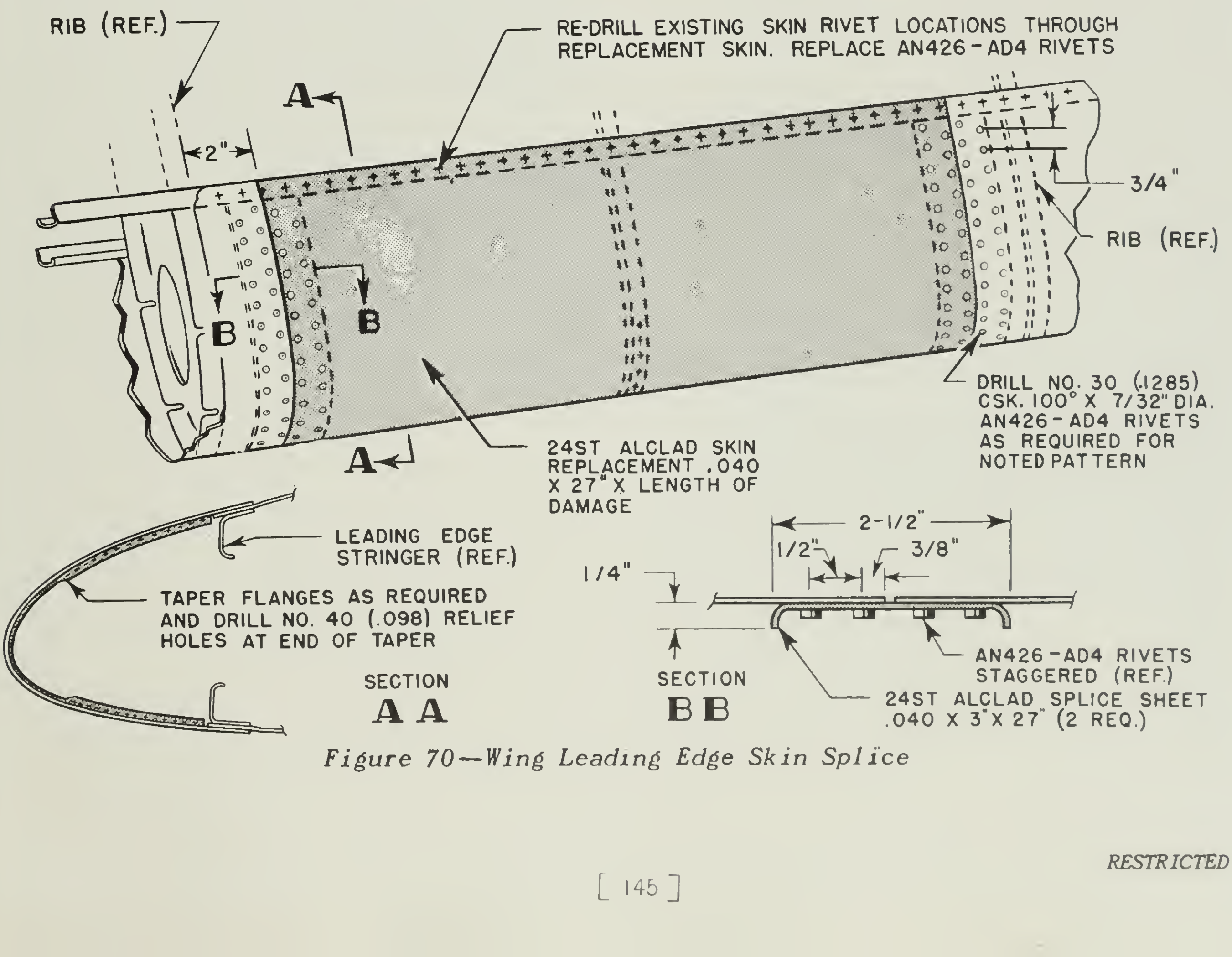


Figure 70—Wing Leading Edge Skin Splice

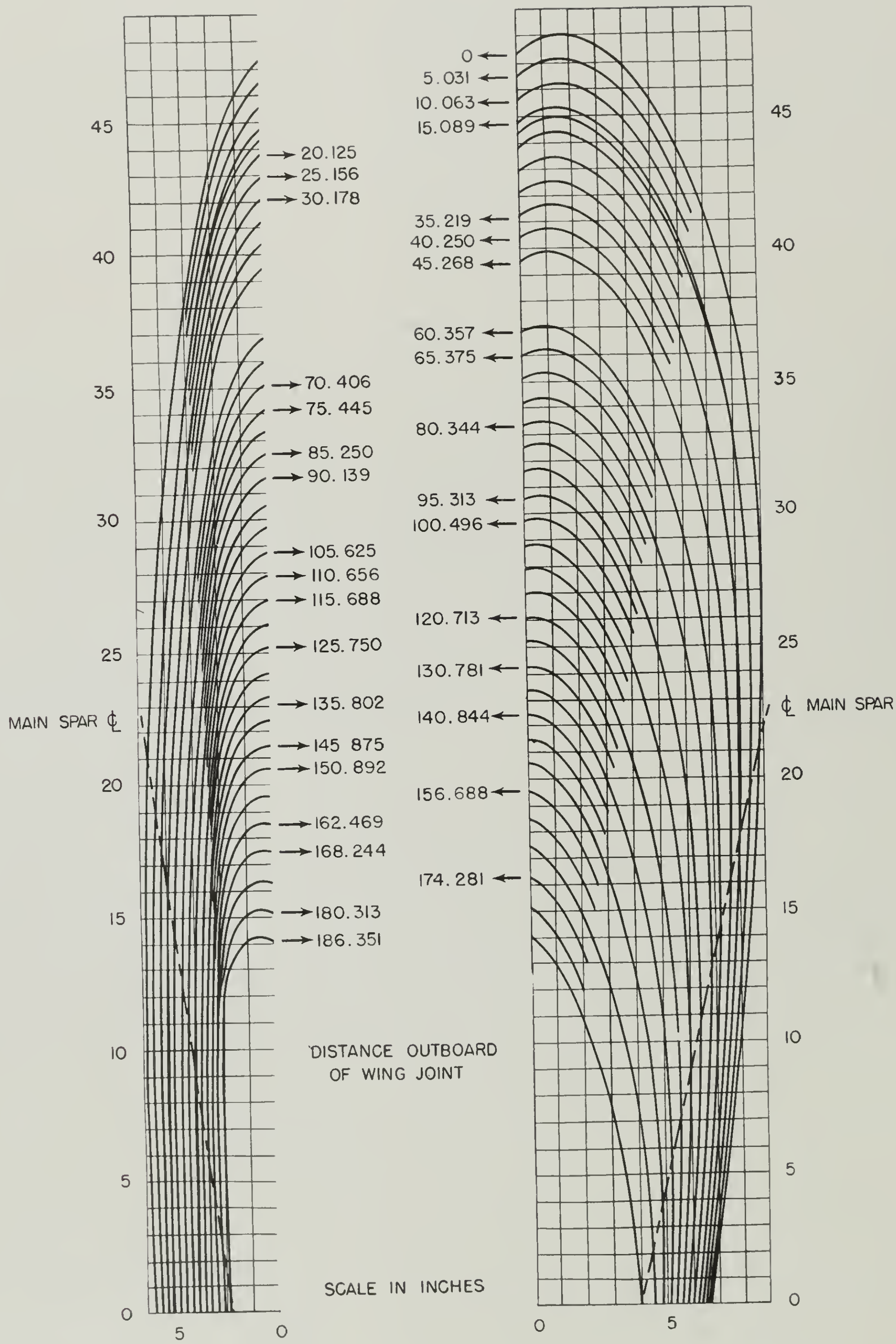


Figure 71—Outer Wing Leading Edge Contours

centers; stagger the rivet locations in each row. Remove the skin panel replacement, burr all the rivet holes, and apply one coat of zinc chromate primer to all overlapping surfaces. Again secure the skin replacement panel to the leading edge with skin fasteners. Rivet the skin panel to the lower leading edge stringer and to the lower areas of the two splice sheets with AN426-AD4 rivets in the quantity required. If it is necessary to gain access, pull the top edge of the replacement panel away from the structure. Next, rivet the up-

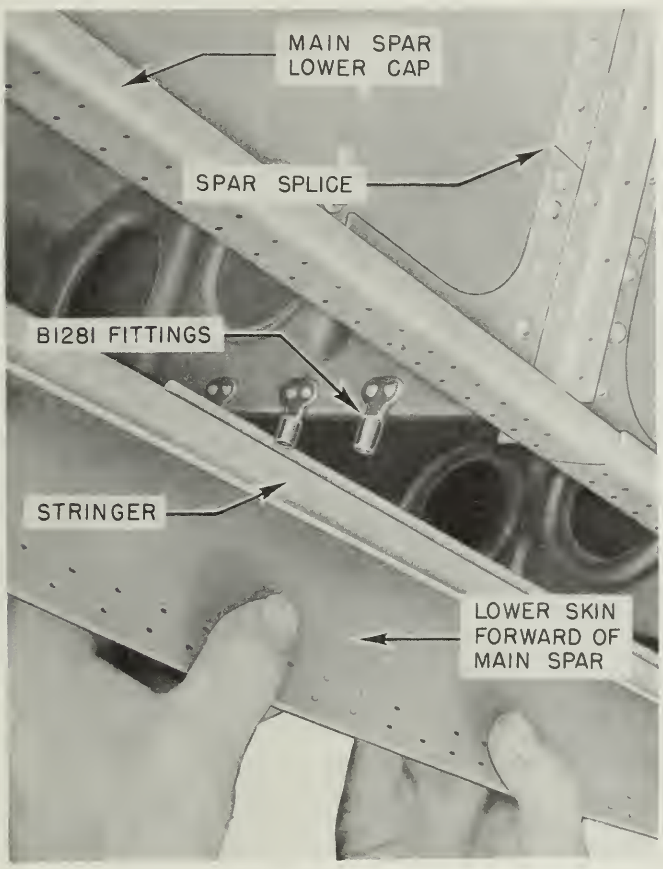


Figure 72—Gaining Access to the Outer Wing Leading Edge

per area of one of the splice sheets and the upper stringer to the skin with AN426-AD4 rivets. Remove the skin fasteners as the drilling progresses. If it is necessary to gain access to buck the rivets, pull the corner of the skin replacement away from the structure. Lastly, rivet the upper area of the skin panel to the remaining splice sheet with AN426-AD4 rivets. If the area is inaccessible for rivet bucking, Cherry blind rivets may be substituted for AN426-AD4 rivets for riveting the upper area of the remaining splice sheet and

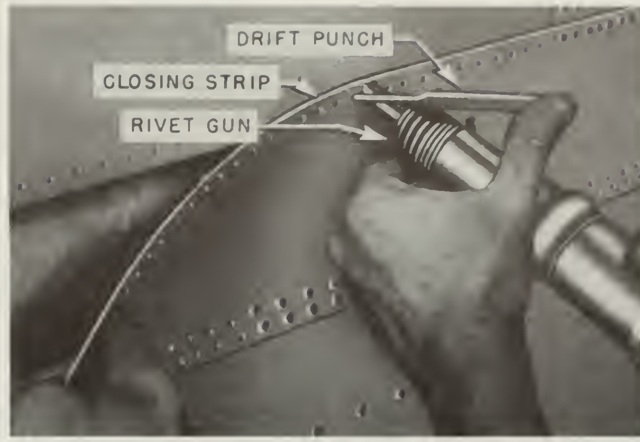


Figure 73—Use of Drift Punch When Riveting Closing Strip Skin

a short length of the stringer. Use LS1126-4-4 Cherry rivets in the quantity required.

54. REMOVING AND REPLACING THE OUTER WING CLOSING STRIP SKIN.

To gain access to the outer wing, in some cases it may be necessary to remove a portion of the narrow closing strip of skin located on the lower surface of the outer wing from root to the tip, just aft of the main spar. Before removing the closing strip, properly support the wing in such a manner as to remove all strain from the structure. If access is desired to the inboard end of the wing, remove the screws located in the bolting angle at the inboard end of the strip; then drill out the rivets securing the strip to the structure. Slide the strip from beneath the bolting angle and curl back as required. If access is desired to the tip of the wing in the vicinity of the closing strip, detach the tip and drill out the rivets securing the strip to the tip structure. Curl

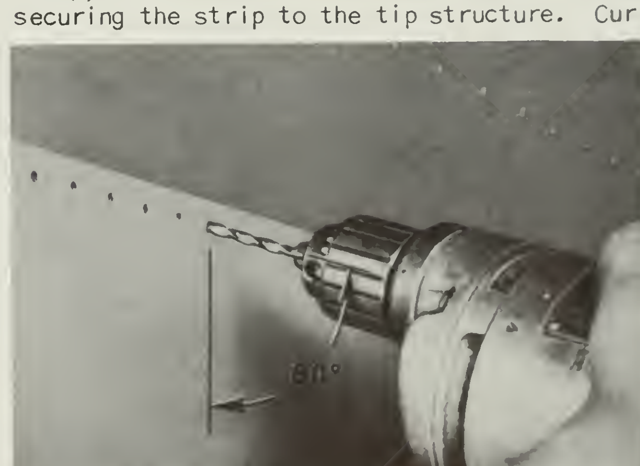


Figure 74—Use of Drill at Closing Strip Skin

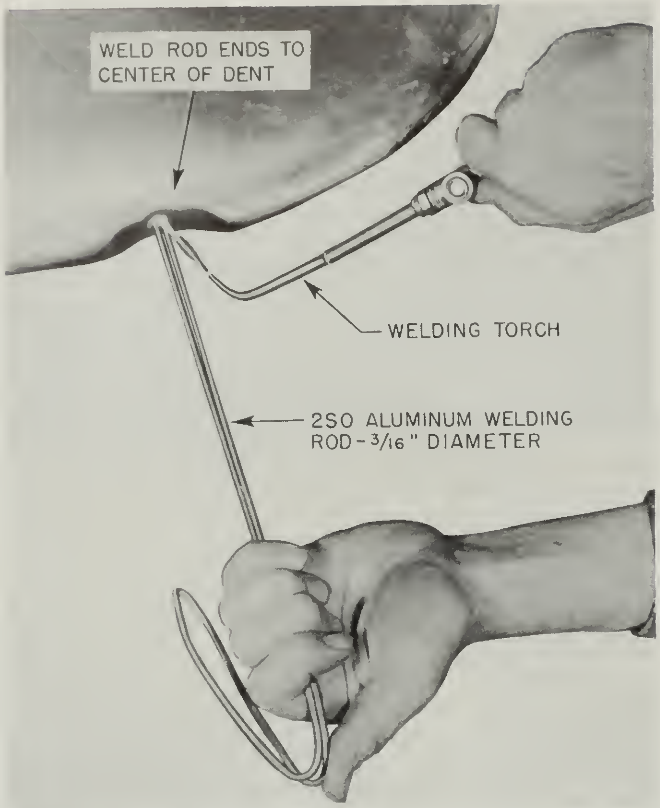


Figure 75--Welding a Rod to a Dent in the Horizontal Stabilizer Tip

back the skin as required. If access is desired to the interior structure forward of the outer wing main spar, the skin panel forward of the closing strip of skin must be removed in certain areas *(see Figure 72)*. When reriveting the strip, prevent the sagging and bulging of the adjacent sheets by inserting a narrow drift punch in the rivet hole just ahead of the rivet being driven. While driving the rivet, tighten the adjacent sheet by pressing the top of the punch forward or aft, as required *(see Figure 73)*. Where the fabrication of a new portion of the closing strip becomes necessary, the general procedure outlined for the removal and replacement of the skin panels is entirely applicable except for the following: Using the damaged section of skin as a template, center punch the hole locations and drill the rivet locations with the size drill required, tilting the drill forward 10 degrees at the rear edge and tilting the drill aft 10 degrees at the front edge of the strip, measured from the normal perpendicular to the skin surface *(see Figure 74)*.

55. REMOVING AND REPLACING SKIN PANELS.

Replacement of an entire panel of skin should be avoided if possible by partial splicing or patching. However, if over one-half the skin panel is damaged, removal and replacement of the complete skin panel may have to be accomplished. If this is the case, carefully drill out all attaching rivets after properly supporting the structure. Use a drill one size smaller than the diameter of the rivet, and place the point of the drill into the depression in the rivet head. If a power tool is used, locate the drill on the rivet first, then apply the power in short bursts. Drill until the rivet head twists from the shank, and then drift the remainder of the rivet free. Particular care must be exercised to prevent the elongation of the existing rivet holes. Remove the damaged skin panel and cut a replacement panel, using the damaged skin as a template if possible. Place the panel beneath the original damaged panel previously removed. Drill through the two panels and temporarily fasten with skin fasteners. Drill most of the skin and stringer rivet locations in the replacement panel, using the existing rivet locations in the original damaged panel as a guide. Install access covers and make other modifications on the replacement skin panel to match the original skin panel.

Figure 76—Removing a Dent in the Horizontal Stabilizer Tip

Burr all the rivet holes and apply one coat of zinc chromate primer to all overlapping surfaces of the replacement skin panel. Place the skin panel in the proper position on the structure and fasten with a sufficient number of skin fasteners to prevent the panel from creeping while riveting. Using the same type and size rivets as were used in the original panel, rivet the replacement panel, first at the corners and then at the intermediate positions.

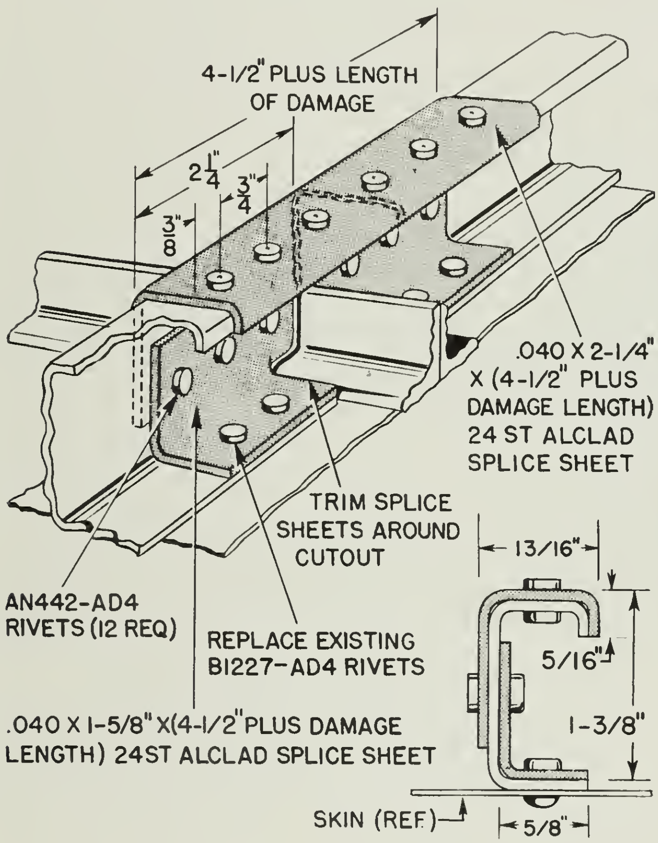


Figure 77—Fuel Tank Compartment Cover Former Splice

56. REMOVING DENTS IN THE HORIZONTAL STABILIZER TIP SKIN.

The tip of the aluminum horizontal stabilizer is formed of weldable 52S-1/2H aluminum alloy, which makes possible the following procedure for removing dents and restoring the contour of the tip. This procedure does not apply to the wing tip, which is formed of nonweldable 24ST alclad. If the damage is in the form of a gradual dent without sharp creases, the method is quite successful; but if the tip is obviously damaged beyond restoration, the tip should be replaced. If repair is to be attempted, cut a 2-foot length of 3/16-inch 2SO aluminum welding rod. Bend up the rod at the center to form a doubled length. Place the ends of the rod at the center of the dent and weld the rod to the dent with a welding torch *(see Figure 75)*. Hydrogen welding facilities are desirable, but oxyacetylene equipment may be used with a tip size slightly larger than would ordinarily be used with steel of the same thickness. Adjust the torch to produce a slightly carbonizing flame, soft and bluish and not "blowy". Use

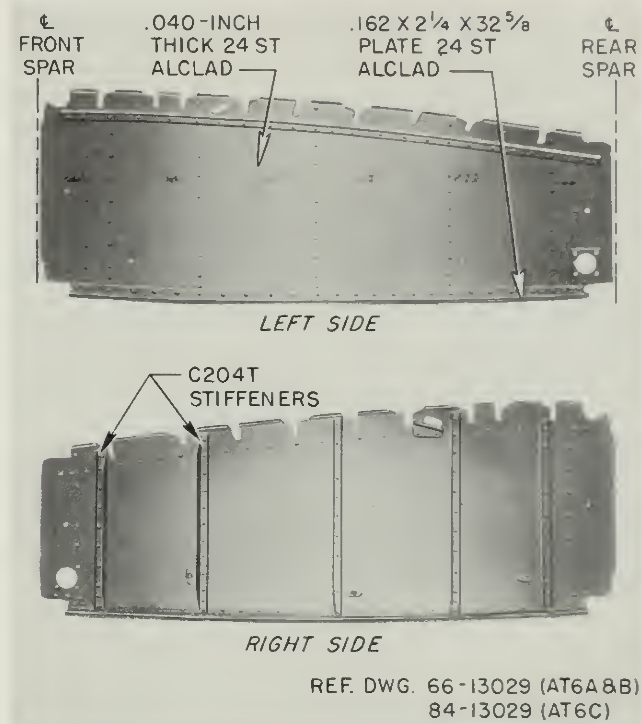


Figure 78—Centersection Intermediate Center Rib

a suitable flux. After the rod is welded to the dent in the tip, carefully heat the surrounding area in the tip and firmly pull on the rod. Remove the torch and continue pulling on the rod until the dent is gradually restored to the proper contour *(see Figure 76)*. Concluding repair, trim the welding rod from the tip.

57. FUEL TANK COMPARTMENT COVER FORMERS.

A typical splice of the fuel tank compartment formers is shown *(see Figure 77)*. If the damage is in the form of a crack, drill a No. 40 (.098) hole at the ends of the crack. For the splice members, two sheets of .040 inch thick 24ST alclad are required. Cut one of the splice members 2-1/2 inches wide and 4-1/2 inches long plus the length of the damage. Cut the other

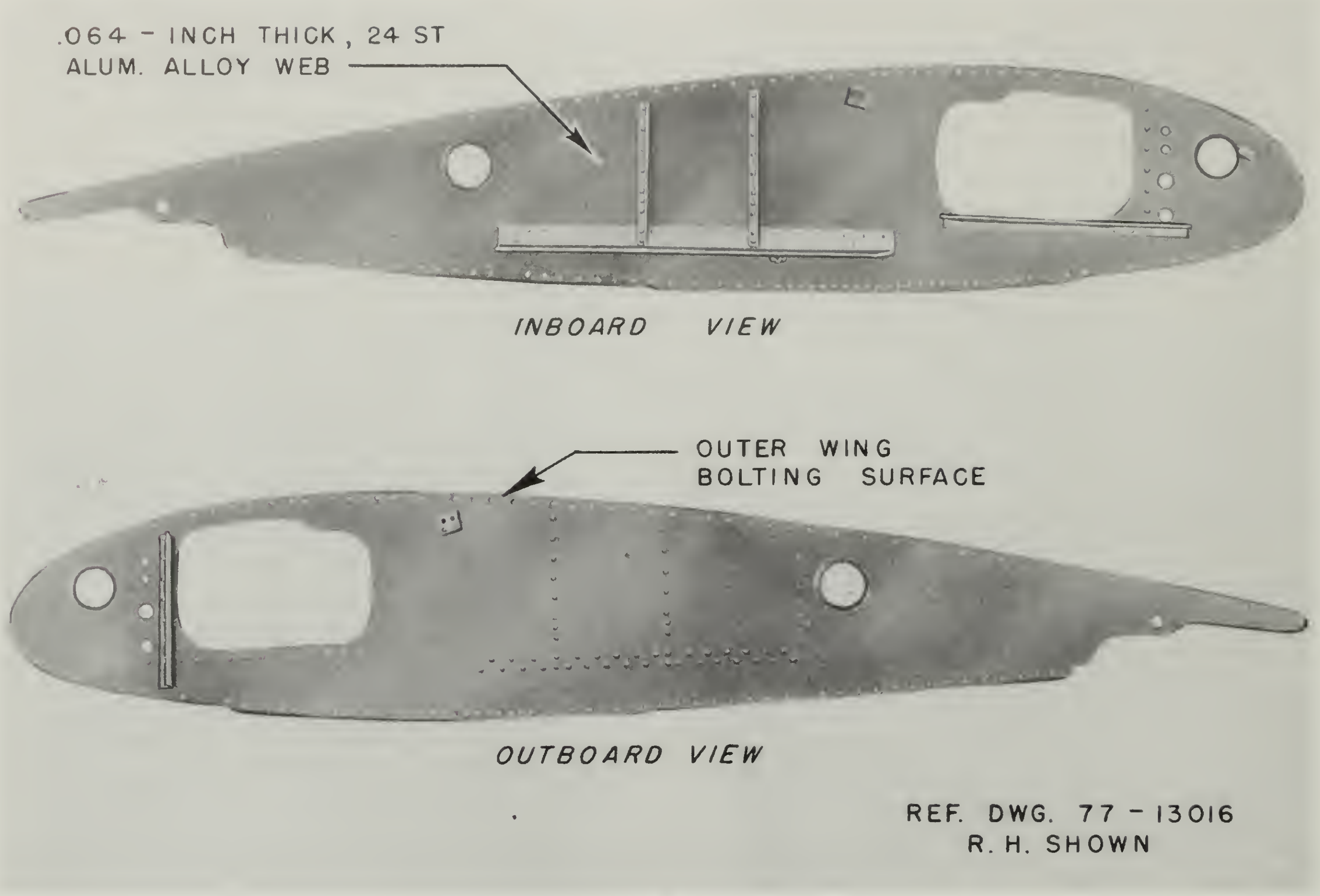


Figure 79 —Centersection End Rib

splice member 1-5/8 inches wide and 4-1/2 inches long plus the length of the damage. Along the length of the 4-1/2 inch wide sheet, bend a 90-degree, 5/16-inch flange; then bend a 90-degree, 13/16-inch flange measured from the first bend. Along the length of the remaining splice sheet, bend a 90-degree, 5/8-inch flange. Trim the splice members around any stringer cutouts in the affected area. Drill out the affected skin rivets at each side of the damage. Clamp the splice sheets to the former, and center punch the required rivet locations at each side of the damage. With a No. 30 (.1285) drill, drill the splice member through the existing rivet holes in the skin, and drill the remaining rivet locations as required. Remove the splice members and burr the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Reclamp the splice members to the former; and through the skin, insert and drive B1227-AD4 rivets in the quantity required. Through the remaining holes, insert and drive twelve AN442-AD4 rivets *(see Figure 77)*.

58. CENTERSECTION INTERMEDIATE RIBS.

The centersection intermediate ribs, located on the upper surface of the centersection between the front spar and the rear spar, may be spliced *(see Figure 80)*. If the damage is extensive, cut out the damaged material with a hack saw, locating the cut between two skin rivets at each side of the damage. Depending upon the extent of the damage, all or part of the complete splice may be used. For example, if only sheet material of the rib is damaged, disregard the extruded member splice and use the splice of the sheet material only. For a complete splice, cut a length of Type K78C extrusion equal to 6 inches plus the length of the damage. Also cut two sheets of .051 inch thick 24ST alclad, each having a width of 2-3/4 inches and a length of 6 inches plus the length of the damage. Along the length of one of the sheet splice members, bend a 90-degree, 5/8-inch flange, observing a minimum bend radius of 5/32-inch. Along the length of the other sheet splice member, bend a 90-degree, 9/16-inch flange. Chamfer the outside corner of the extruded K78C splice member to fit the inside radius of the existing extruded flange on the rib. With a No. 40 (.098) drill, drill out the affected skin rivets at each side of the damage. Clamp the splice members to the proper position on the rib. With a No. 30 (.1285) drill, drill through the extruded splice member, using the existing rivet holes in the rib as a guide. Center punch and drill new rivet locations through the sheet splice members as required to obtain the noted pattern. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the sheet splice members and to all surfaces of the extruded member. Again clamp the splice members to the rib. Through the skin rivet holes, insert and drive B1227-AD4 rivets in the quantity required. Through the remaining twenty-six holes in the rib, insert and drive AN442-AD4 rivets *(see Figure 80)*.

59. WING TRAILING EDGE RIBS AND CENTERSECTION LEADING EDGE RIBS.

The wing trailing edge ribs, inboard of the aileron cutout, and the centersection leading edge ribs may be spliced as shown *(see Figure 81)*. Replacement of these ribs may be very readily accomplished; but if spare parts are not available, it is entirely acceptable to splice the rib. If the damage is in the form of a crack, drill a No. 40 (.098) hole at the ends of the crack. If the damage is extensive, cut out the damaged material between skin rivets. For the splice members, bend up two sheets of .040 inch thick 24ST alclad having a typical width of 3 inches and a length of 2-1/2 inches plus the length of the damage. Along the length of one of the sheets, bend up a 90-degree, 1/2-inch flange. Along the length of the other splice member, bend a 90-degree, 5/8-inch flange. Trim the splice sheets as required. With a right-angled drill, drill out the affected skin rivets at each side of the damage. Clamp the splice sheets to the rib and center punch two staggered rivet rows at each side of the damage. On the lower rib flange, center punch two rivet locations at each side of the damage. With a No. 30 (.1285) drill, drill the center-punched rivet locations and drill the splice members through the skin rivet holes on the

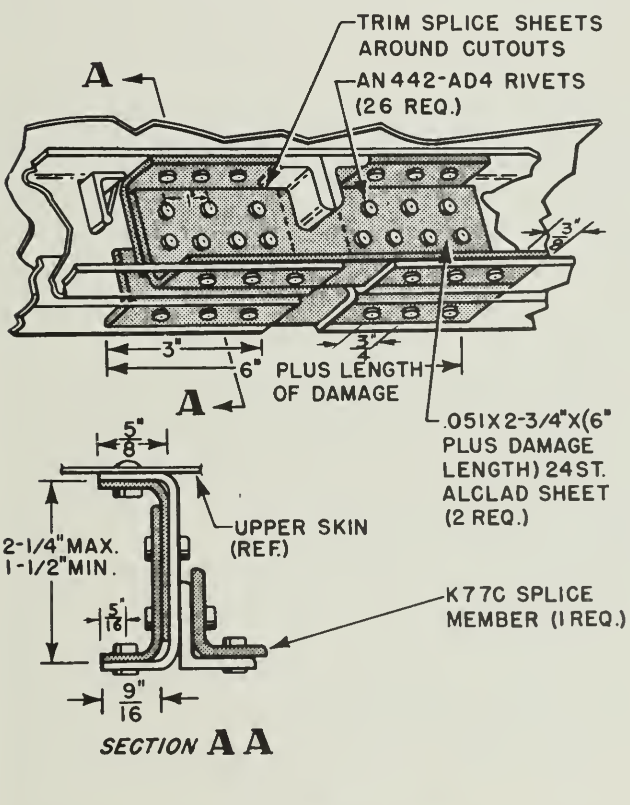


Figure 80—Centersection Intermediate Rib Splice

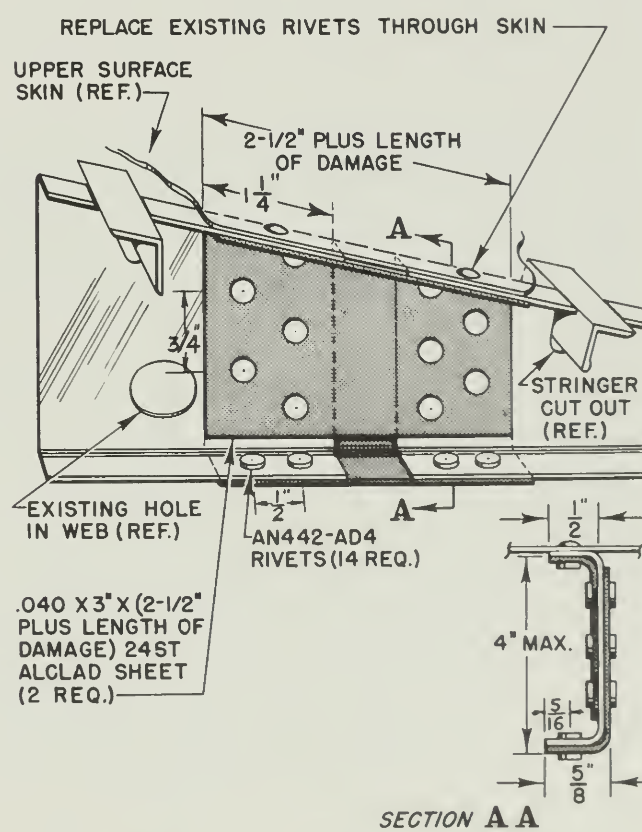


Figure 81—Trailing Edge Rib Splice

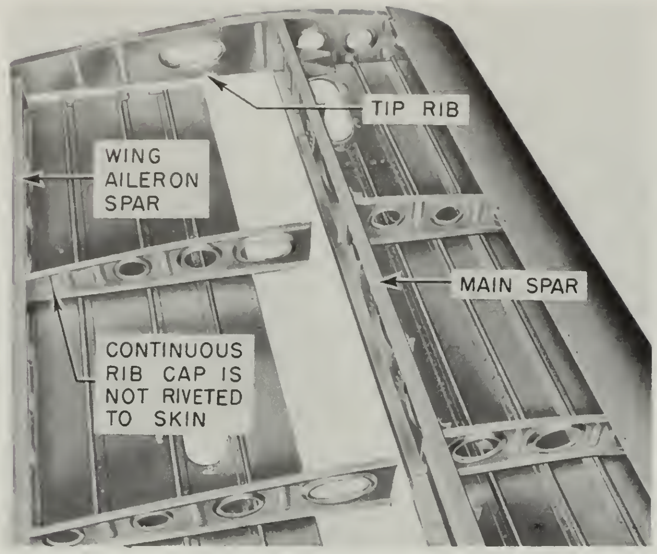


Figure 82—View of Outer Wing Section Showing Ribs Installed

upper cap of the rib. Remove the splice members and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Again clamp the splice members to the rib; and through the skin rivet holes, insert and drive B1227-AD4 rivets in the quantity required. Through the remaining rivet holes, insert and drive AN442-AD4 rivets in the quantity required to obtain two vertical rivet rows at each side of the damage *(see Figure 81)*.

60. DAMAGED RIB BEADS.

The repair of damage located on and about beads in the rib web is shown *(see Figure 84)*. If the damage is in the form of a crack, drill a No. 40 (.098) hole at the ends of the crack. For the reinforcement, bend up a section of .040 inch thick 24ST alclad having a typical dimension of 2-5/8 x 1-1/2 inches. Along the vertical edges of the reinforcement, bend up 90-degree, 1/4-inch flanges. Observe a minimum bend radius of 1/8-inch. Apply the reinforcement to the rib side opposite the bead. Center punch the four rivet locations and drill through the reinforcement with a No. 30 (.1285) drill. Remove the reinforcement and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces and again secure the reinforcement to the rib. Into the rivet holes, insert and drive four AN442-AD4 rivets *(see Figure 84)*.

61. DAMAGED RIB CUTOUTS.

The repair of damage located on and about

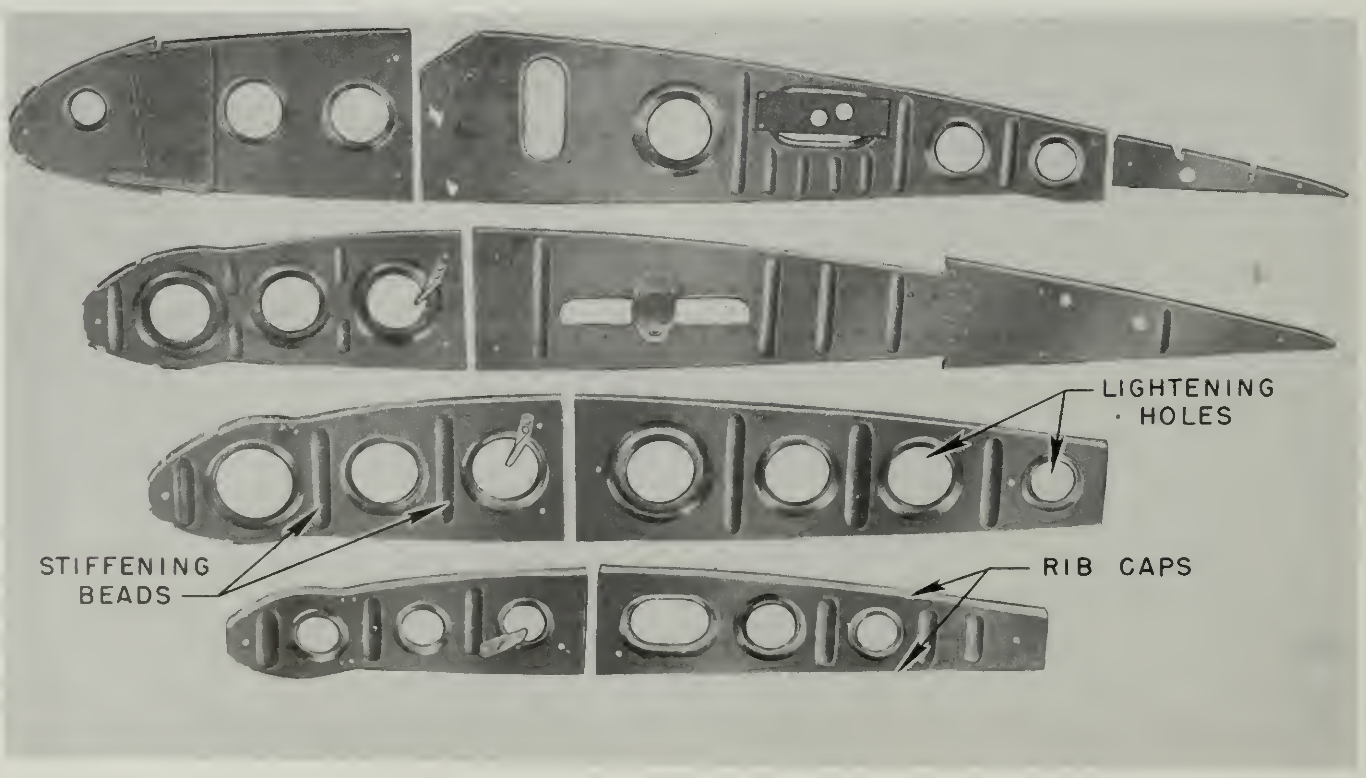


Figure 83—Typical Ribs

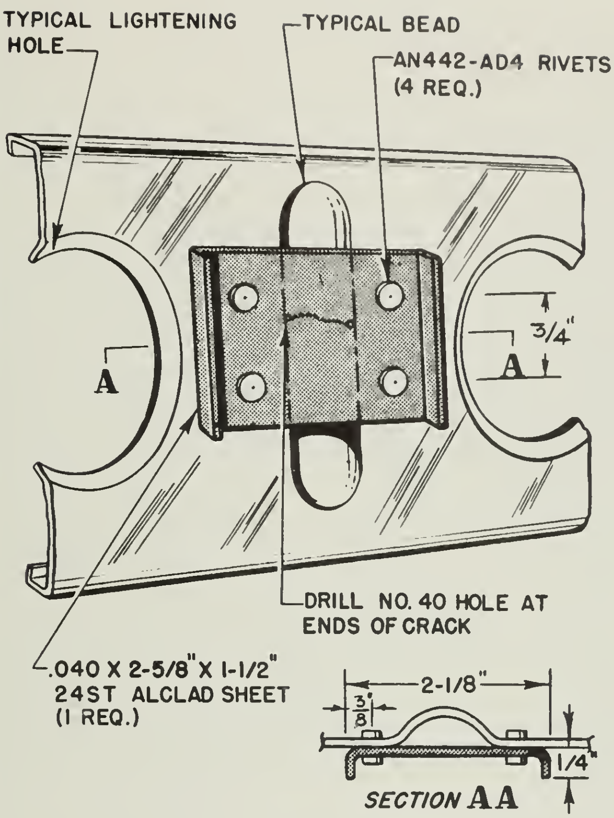


Figure 84—Typical Repair of Broken Rib Beads

lightening cutouts in the rib webs is shown *(see Figure 85)*. If the damage is in the form of a crack, drill a No. 40 (.098) hole at the end of the crack. For the reinforcement, bend up a section of .040 inch thick 24ST alclad having a typical dimension of 3-3/4 x 1-3/4 inches. Along the length of one of the edges, bend up a 90-degree, 1/4-inch flange. Observe a minimum bend radius of 1/8-inch. Apply the reinforcement to the rib side opposite the cutout flange. Center punch the nine rivet locations and drill through the reinforcement with a No. 30 (.1285) drill. Remove the reinforcement and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces and secure the reinforcement to the rib. Into the rivet holes, insert and drive nine AN442-AD4 rivets *(see Figure 85)*.

62. BUCKLED RIB WEBS.

Buckled rib webs that are not cracked or distorted beyond restoration may be straightened and repaired by the application of an angle perpendicular to the rib flanges *(see Figure 86)*. For the reinforcement, bend up a sheet of .040 inch thick 24ST alclad having a typical dimension of 4 x 1-1/4 inches. Depending upon the extent of the damage, one or several reinforcing angles may be used in the affected area. Cut off the legs of the angles to a gradual taper, as shown. Apply the reinforcement to the rib web and securely clamp in place to restore the shape of the rib web. Center punch four rivet locations and drill through, using a No. 30 (.1285) drill. Remove the reinforcement, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces, and again clamp the reinforcement to the rib web. Into the rivet holes, insert and drive AN442-AD4 rivets *(see Figure 86)*.

63. DAMAGED RIB FLANGES.

With regard to rib flanges, there are two types of ribs employed in the construction of the fixed surfaces; one type having the rib flanges riveted to the skin and cut around the stringers, and the other type having the unbroken flange located approximately 3/4-inch within the skin mold line. The repair of damaged rib flanges riveted to the skin is shown *(see Figure 87)*. The repair of damaged rib

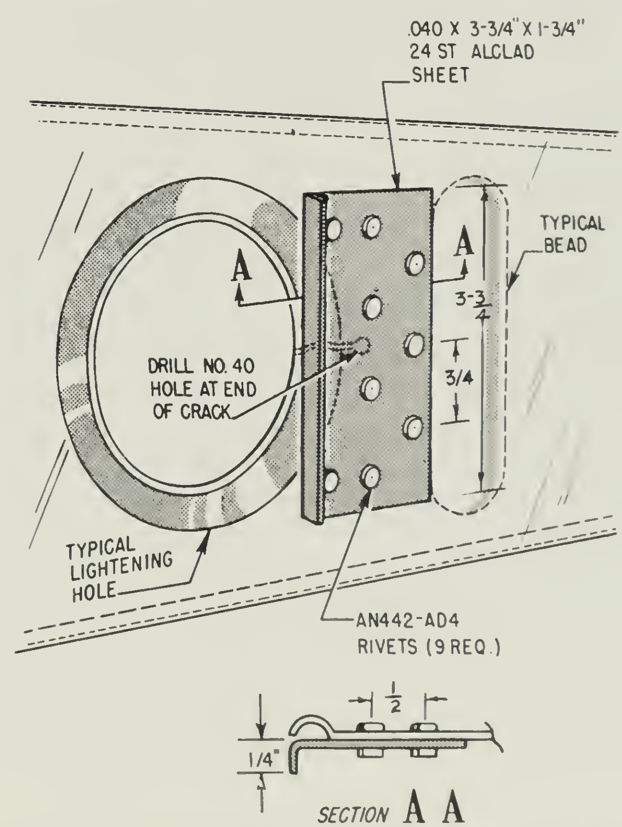


Figure 85—Typical Repair of Broken Rib Cutouts

flanges not riveted to the skin is also shown *(see Figure 88)*. These repairs may be used on bent-up ribs up to and including .064 inch thick alclad. The procedure is as follows: Drill a No. 40 (.098) hole at the end of the break in the rib flange. Where the rib flange is riveted to the skin, prepare a reinforcing angle of 24ST alclad sheet of the same thickness as the rib, and cut the angle of sufficient length to extend between adjacent stringers. Where the rib flange is not riveted to the skin, cut a sheet of 24ST alclad of the same thick-

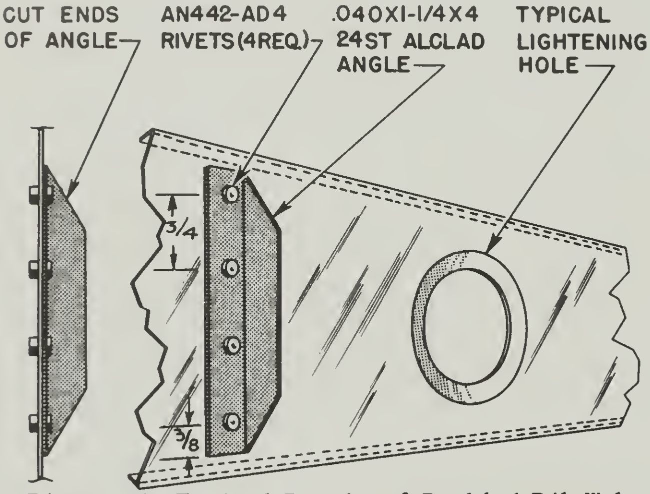


Figure 86—Typical Repair of Buckled Rib Webs

ness as the rib and of a length sufficient to pick up four rivets at each side of the break. Bend up the sheet to fit the rib flange on the outside. Drill out the affected rivets at each side of the damage and fit the reinforcing angle to the rib cap. Where the rib flange is riveted to the skin, drill the angle through, using the same size drill as originally used through the skin. Where the rib flange is not riveted to the skin, use a No. 40 (.098), a No. 30 (.1285), and a No. 21 (.159) drill to drill the rivet holes through .020-, .025-, and .051-inch rib material, respectively *(see Figure 88)*. Remove the angle and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces and resecure the angle to the rib flange. Where the rib flange is riveted to the skin, replace the skin rivets; and through the rib web and angle, insert and drive AN442-AD rivets of the same size as the existing skin rivets. Where the rib flange is not riveted to the skin, use AN442-AD3 rivets through .020 inch thick alclad, AN442-AD4 rivets through .025 inch thick material, and AN442-AD5 rivets through .051 inch thick alclad. Use this repair only on bent-up ribs where there are no doublers or other reinforcements within 8 inches of the damaged flange.

64. SPLICING TYPICAL RIBS.

The centersection ribs between the rear spar and the flap spar, and all the outer wing nose ribs and intermediate ribs may be spliced as outlined herein. Cut out the damaged rib material with a hack saw, locating the cut at the center of a lightening hole *(see Figure 89)*. For the splice members, cut two sheets of .040 inch thick 24ST alclad having a width of 2-1/4 inches and a length of 4-1/2 inches plus the length of the damage. Along one of the edges of the length of each of the splice sheets, bend a 90-degree, 1/4-inch flange and then bend a 90-degree, 5/8-inch flange measured from the first bend. Along the remaining edge of the length of each of the splice sheets, bend a 90-degree, 3/16-inch flange and then bend a 90-degree, 1/2-inch flange measured from the first bend. Observe a minimum bend radius of 1/8-inch. Clamp the splice members to the upper and the lower caps of the rib, and center punch the twenty-eight required rivet locations. With a No. 30 (.1285) drill, drill the center-punched rivet locations. Remove the splice members and burr all the rivet holes. Apply

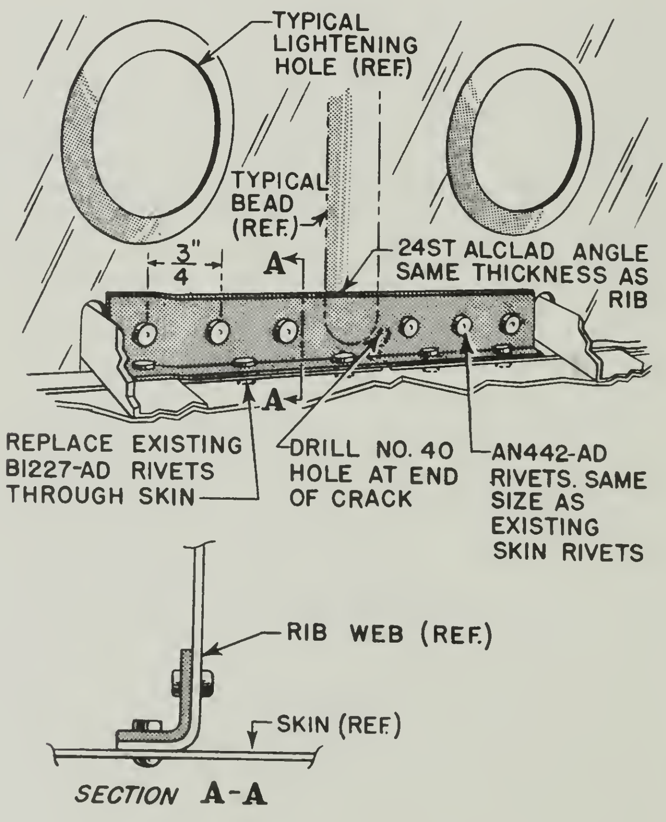


Figure 87—Typical Rib Flange Repair Where Flange Is Riveted to Skin

one coat of zinc chromate primer to all overlapping surfaces. Again clamp the splice members to the rib, and insert and drive twenty-eight AN442-AD4 rivets into the rivet holes.

65. REMOVING AND REPLACING DAMAGED RIBS.

Local investigation will determine the procedure to follow in gaining access to a damaged rib. With a No. 40 (.098) drill, drill out the rivets securing the rib flange to the skin and the stringer clips. Replacement of the ribs should be accomplished by utilizing North American Aviation, Inc., spare parts *(see Figures 1 and 2)*. Where spare parts are not available and immediate repair is necessary, use the damaged rib as a template and fabricate a replacement locally. Lay out the shape of the affected rib upon a panel of 1/2-inch wood. In several places, temporarily nail this panel to a second panel of 1/2-inch wood; and with a keyhole saw, cut around the marked contour. *(See Figure 90)*. Pull out the nails and separate the two equivalent panels cut to the contour of the affected rib. Measure the developed width of the rib flange and cut an equivalent sheet of 24ST alclad equal to the

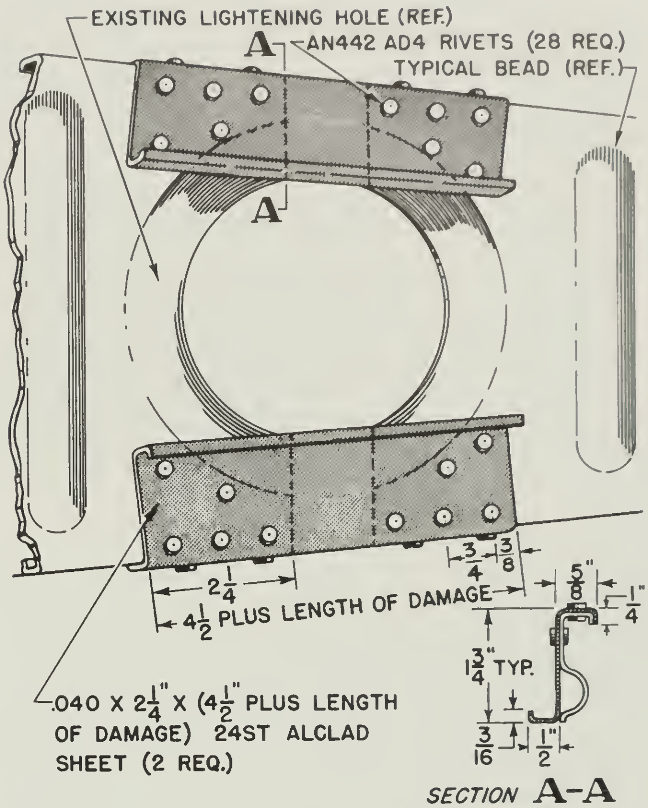


Figure 89—Typical Rib Splice

contour of the wood panels plus the developed width of the rib flange all around. Clamp the sheet of alclad between the two wood form blocks in a vise; and with a mallet, pound the flanges of the replacement rib over the edge of one of the wood form blocks *(see Figure 91)*. Using the damaged rib as a guide, bend up the same type of flanges as were originally employed. In place of the flanged cutouts and stiffening beads employed on the original rib, bend up 1-1/4 inch wide strips of .040 inch thick 24ST alclad to a right angle, and secure them on the rib web perpendicular to the rib flanges. Taper cut the upstanding leg of the stiffener as shown *(see Figure 86)*. Install the replacement rib in exactly the same manner as the original.

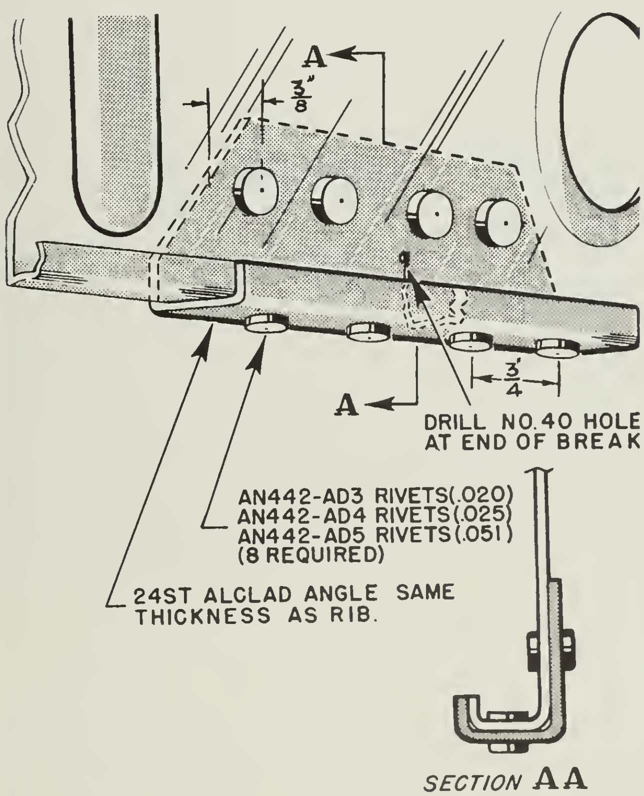


Figure 88—Typical Rib Flange Repair Where Flange Is Not Riveted to Skin

66. WING JOINT BOLTING ANGLES.

No repairs to the wing joint bolting angles are permissible. Replacement of damaged bolting angles must be accomplished. The centersection bolting angle consists of four sections *(see Figure 92)*. The outer wing bolting angle also consists of four sections *(see Figure 93)*. Disassemble the wing at the joint, and drill out the existing rivets through the

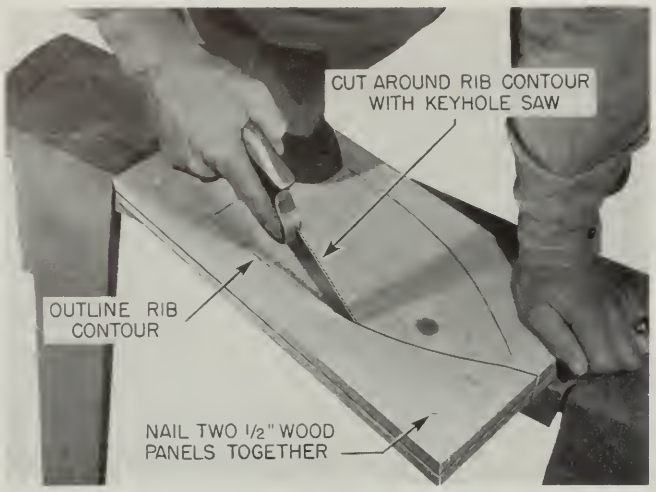


Figure 90—Cutting Replacement Rib Form Blocks

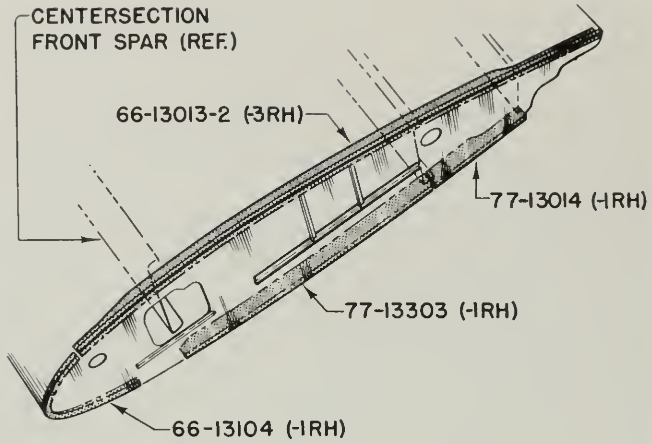


Figure 92—Centersection Bolting Angle Part Numbers

bolting angle section and the skin, using extreme care to prevent the elongation of any of the existing rivet holes. Use a drill one size smaller than the rivet diameter. Apply the drill to the depression in the rivet head and drill in short bursts until the rivet head twists from the shank. Punch out the remaining portion of the rivet. Use this procedure to remove all the rivets from the bolting angles. Remove the screws from the remaining holes and take off the damaged angle. The fuel tank cover bolting angle and the lower rear bolting angle of the centersection are supplied as spare parts with the pilot holes drilled in the proper locations. The top bolting angle of the centersection and all of the bolting angles of the outer wing are supplied as spare parts without pilot holes. In either case, line up the new bolting angle flush with the skin edge, and securely clamp the angle in position with heavy "C" clamps. On the new bolting angles without drilled pilot holes, drill through the bolting angle with a No. 40 (.098) pilot drill, using the existing rivet holes in the skin as a guide. Then drill the pilot holes through with a drill equal to the size of the existing holes in the skin. As the drilling progresses, slip a thin hook-shaped hack saw blade, with the teeth removed, between the bolting angle and the skin and scrape out the drill chips. Using an undamaged wing for reference, countersink and spot-face

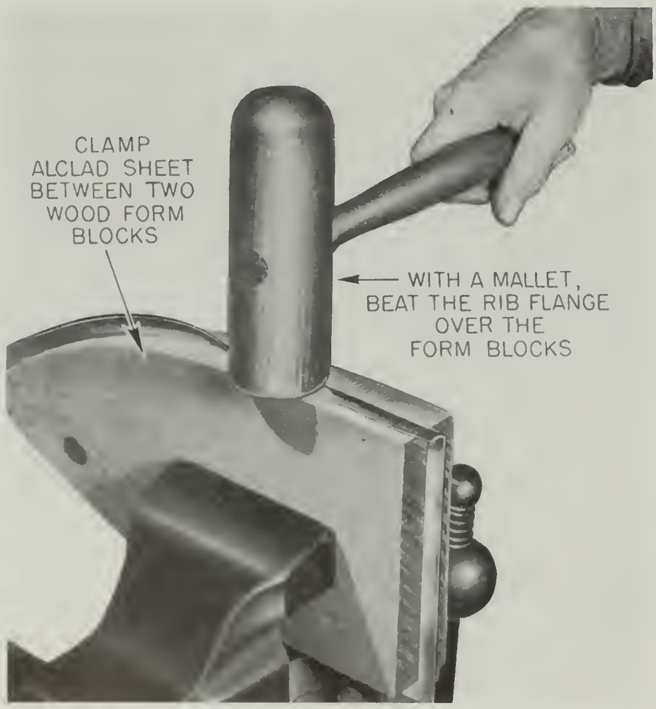


Figure 91—Forming Replacement Rib

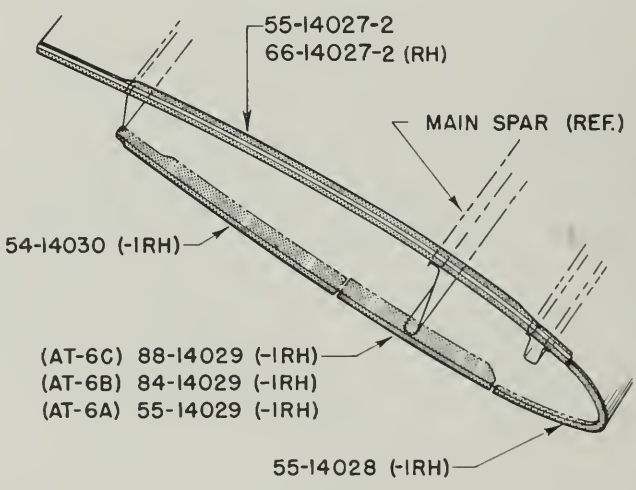


Figure 93—Outer Wing Bolting Angle Part Numbers

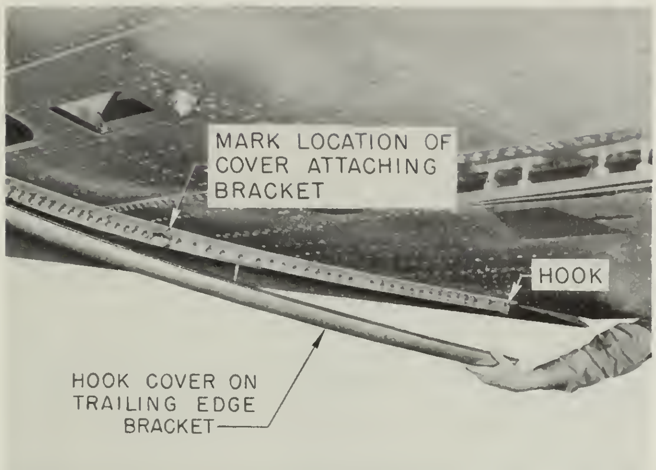


Figure 94—Installing Lower Portion of Wing Joint Cover

the various holes. Again using an undamaged wing as a guide, insert and drive the required size and type rivets and install the required screws.

67. WING JOINT COVERS.

Seriously damaged wing joint covers must be replaced rather than repaired because the alignment of the covers is critical. A splice would affect the alignment of the cover and prevent proper installation. For replace-

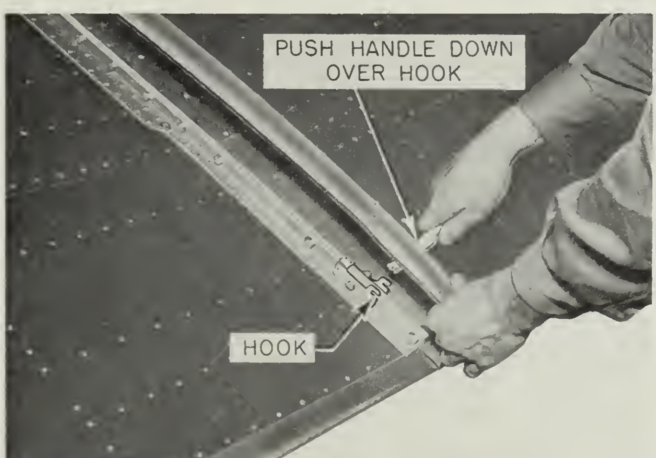


Figure 95—Installing Upper Portion of Wing Joint Cover

ments, North American Aviation, Inc., spare parts must be used. Obtain the NA 54-10020 lower cover and the NA 77-10023 upper cover, and assemble them by means of two B1286-8-7 screws at the leading edge portion *(see Figure 97)*. On the lower outboard surface skin,

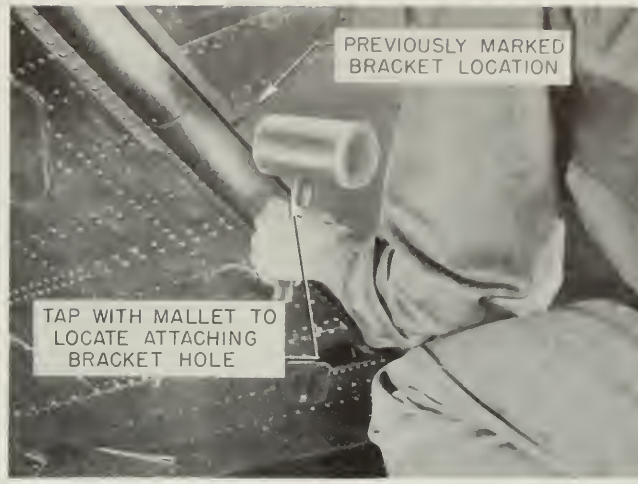


Figure 96—Locating Bracket Hole in Wing Joint Cover

make a grease pencil mark at the centerline of the screw hole in the wing cover attaching bracket *(see Figure 94)*. Cut off the head of a B1286-8-7 screw, file the end down to a point, and thread the screw into the wing cover attaching bracket hole, leaving the sharp point protruding about 1/8-inch. This serves as a back center punch to locate the hole through the cover. Slip the cover assembly over the leading edge of the wing joint. Grasp the lower portion of the cover, and hook it over the small bracket at the lower trailing edge of the wing bolting angles *(see Figure 94)*. Continue

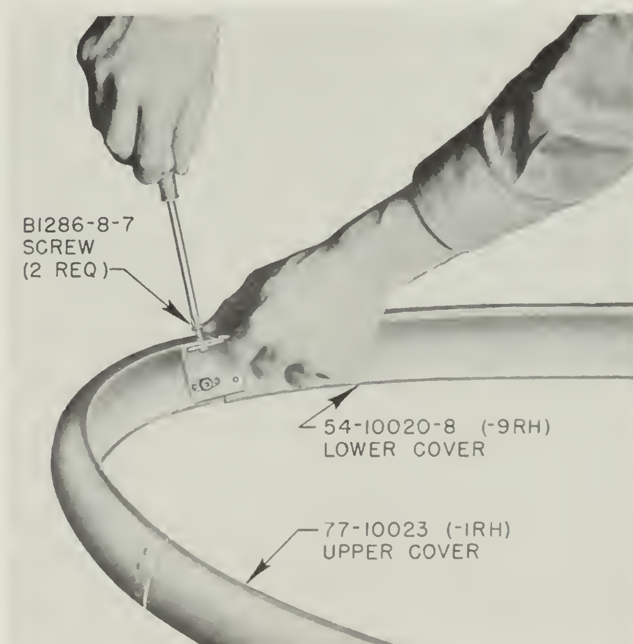


Figure 97—Assembling Wing Joint Cover

holding the lower trailing edge of the cover to keep the cover hooked; then grasp the upper trailing edge of the cover and lift up the handle. Pull aft on the upper cover; and when the cover is in position, push the handle down over the hook to complete the installation *(see Figure 95)*. Along the lower surface skin where the centerline of the screw hole in the cover attaching bracket is marked, measure 7/8-inch down on the contour of the cover from the edge of the cover, and tap the cover lightly with a mallet *(see Figure 96)*. The previously installed sharp screw protruding from the bracket screw hole will center punch the hole location on the inside of the cover. Remove the cover; and in the cover, drill the center-punched rivet location with a 3/8-inch diameter drill. Again install the cover as outlined above. Through the bracket hole that was just drilled in the cover, install a B1286-8-7 screw through a B985-11-20-32 washer. This completes the installation.

68. EXTRUDED ALUMINUM REPAIR MATERIAL.

The various extruded cross-sectional shapes that are required in the repair of the structural members of the fixed surfaces are listed below. The material used in forming these extrusions is 24ST aluminum alloy, conforming to Federal Specification QQ-A-354. Numbers with a C prefix are North American Aviation, Inc., standard parts. Following the North American Aviation, Inc., part number, the Aluminum Company of America (Alcoa) equivalent die number is listed in parentheses.

EXTRUSION DIE NUMBERS

C148T (K734-JJ)	C265T (L23557)
C180T (K14280)	C266T (L23558)
C203T (K14653)	C274T (L23966)
C204T (K14654)	C366T (L24910)
C245T (K16862)	K77A (K77A)
C250T (K16869)	K77C (K77C)

69. ALCLAD SHEET REPAIR MATERIAL.

The aluminum sheet material used in the repair of the wing should be 24ST alclad conforming to Federal Specification QQ-A-362. The various thicknesses of sheet that may be required are listed below. The lengths and widths of the sheets must be determined locally for the extent of repairs to be undertaken, with reference to the repair procedures outlined in this Section.

THICKNESS OF 24ST ALCLAD (INCHES)	REMARKS
.020	RIBS, SKIN (C107LT-20)
.025	RIBS, SKIN (C373LT)
.032	RIBS, SKIN, SPARS
.040	RIBS, SKIN, SPARS (C123LT)
.051	RIBS, SKIN, SPARS
.064	RIBS, SKIN, SPARS

70. RIVETS, BOLTS, SCREWS, AND NUTS REQUIRED FOR REPAIR.

The following types of rivets, bolts, screws, and nuts may be required for the repair of the fixed surfaces. Types of rivets, bolts, screws, and nuts are described in Section I.

PART NO.	DESCRIPTION	
B1227-AD3	RIVET, BRAZIER HEAD	3/32 DIA.
B1227-AD4	RIVET, BRAZIER HEAD	1/8 DIA.
B1227-AD5	RIVET, BRAZIER HEAD	5/32 DIA.
B1227-AD6	RIVET, BRAZIER HEAD	3/16 DIA
AN426-AD3	RIVET, 100° C'SUNK	3/32 DIA.
AN426-AD4	RIVET, 100° C'SUNK	1/8 DIA.
AN426-AD5	RIVET, 100° C'SUNK	5/32 DIA.
AN426-AD6	RIVET, 100° C'SUNK	3/16 DIA.
AN442-AD3	RIVET, FLAT HEAD	3/32 DIA.
AN442-AD4	RIVET, FLAT HEAD	1/8 DIA.
AN442-AD5	RIVET, FLAT HEAD	5/32 DIA.
AN442-AD6	RIVET, FLAT HEAD	3/16 DIA.
LS-1127-4-2	RIVET, CHERRY BLIND BRAZIER	1/8 DIA.
LS-1127-4-4	RIVET, CHERRY BLIND BRAZIER	1/8 DIA.
LS-1127-4-6	RIVET, CHERRY BLIND BRAZIER	1/8 DIA.
LS-1127-4-8	RIVET, CHERRY BLIND BRAZIER	1/8 DIA.
LS-1127-5-2	RIVET, CHERRY BLIND BRAZIER	5/32 DIA.
LS-1127-5-4	RIVET, CHERRY BLIND BRAZIER	5/32 DIA.
LS-1127-5-6	RIVET, CHERRY BLIND BRAZIER	5/32 DIA.
LS-1127-5-8	RIVET, CHERRY BLIND BRAZIER	5/32 DIA.
LS-1127-6-4	RIVET, CHERRY BLIND BRAZIER	3/16 DIA.
LS-1127-6-6	RIVET, CHERRY BLIND BRAZIER	3/16 DIA.
LS-1127-6-8	RIVET, CHERRY BLIND BRAZIER	3/16 DIA.
LS-1126-4-4	RIVET, CHERRY BLIND 100° C'SUNK	1/8 DIA.
B1251-1032-14	SCREW, NO. 10	
B1251-1032-17	SCREW, NO. 10	
B1251-428-14	SCREW,	1/4 DIA.
B1248-1032-17	SCREW, NO. 10	
B1083	NUT, ELASTIC STOP	1/4 DIA.
AC365-1032	NUT, NO. 10	
22GCH-048	CHANNEL, BOOTS NUT	
22G1-048	NUTS, BOOTS	

71. REPAIR TOOLS.

The following list is a brief summation of the tools that are required in repairing the fixed surfaces:

TOOL	REMARKS
BARS, RIVET BUCKING	SEE SECTION I FOR TYPES OF BARS.
BRAKE, HAND	USED FOR BENDING ALCLAD SHEET.
BURNISHING TOOL	USED TO SMOOTH MINOR SCRATCHES IN METAL.
BURRING TOOL	USED TO SMOOTH TRIMMED EDGES OF METAL.
CALIPERS, SLIDE	INSIDE OR OUTSIDE DIAMETER MEASUREMENT.
CHISEL	USED FOR METAL CUTTING AND REMOVAL OF RIVET HEADS WHERE DRILLING IS IMPRACTICAL.
CLAMP, "C"	USED FOR HOLDING TWO OR MORE PIECES OF METAL UNTIL DRILLING AND/OR RIVETING IS ACCOMPLISHED.
CLAMP, SKIN	USED TO SECURE METAL WHEN RIVETING, DRILLING, ETC.
COUNTERSINK	COUNTERSINKS HOLES IN HEAVY STOCK FOR INSERTION OF COUNTERSUNK RIVETS AND SCREWS.
COUNTERSINK, BACK	USED FOR COUNTERSINKING IN LOCATIONS WHERE MOTIVE POWER CANNOT BE APPLIED ON SIDE DESIRED.
DIVIDERS	USED FOR SHEET METAL LAYOUT.
DRILL, HAND	USED FOR RESTRICTED OR SLOW DRILLING, AND ALSO IN THE PRESENCE OF GAS FUMES, IF AN AIR MOTOR IS NOT AVAILABLE.
DRILLS, TWIST	THESE DRILLS USED WITH HAND OR POWER DRILLS ARE STANDARD.
EXTRACTOR, EZY-OUT	USED FOR REMOVING THE REMAINING PORTIONS OF BROKEN SCREWS, BOLTS, AND PINS. DRILL HOLE IN OBJECT APPROXIMATELY ONE-HALF THE DIAMETER, INSERT EZY-OUT, AND WITHDRAW OBJECT BY COUNTERCLOCKWISE ROTATION.

TOOL	REMARKS
FASTENERS, SKIN	USED TO SECURE METAL WHEN RIVETING, DRILLING, ETC.
FILES, HAND	USED FOR FINISHING WORK ON ALL TYPES OF METAL, PLASTIC, AND FIBER, AND ENLARGING HOLES, BURRING EDGES, SMOOTHING CURVES, REMOVING METAL, AND SHARPENING TOOLS.
FILE HOLDER	USED TO HOLD FILES IN CURVED POSITION.
GUN, CHERRY RIVET	MODELS G10 OR G15 *(SEE SECTIO! IX).*
GUN, CLECO SAFETY	USED TO INSERT SKIN FASTENERS.
GUN, PNEUMATIC	USED TO DRIVE STANDARD RIVETS.
HACK SAW, KEYHOLE	USED FOR CUTTING HOLES IN RESTRICTED PLACES.
HACK SAW, LARGE, ADJUSTABLE	USED FOR METAL, FIBER, AND PLASTIC CUTTING. BLADES VARY IN LENGTH AND IN NUMBER OF TEETH PER INCH.
HACK SAW, SMALL	USED FOR LIGHT CUTTING OF METAL FIBER, OR PLASTIC.
HAMMER, FINISHING	USED TO FORM AND FINISH METAL.
MALLET, LEAD	USED FOR SHEET METAL FORMING.
MALLET, PARALYN, RAWHIDE	USED TO FORM AND BEND METAL.
MALLET, RUBBER	USED FOR SHEET METAL LAYOUT.
PILOT, HOLE SAW	USED TO PILOT HOLE SAWS.
PLIERS, CLAMPING	USED FOR HOLDING TWO OR MORE PIECES OF METAL UNTIL DRILLING AND/OR RIVETING IS ACCOMPLISHED.
PROTRACTOR, FLAT	USED FOR SHEET METAL LAYOUT.
PUNCHES, CENTER	USED FOR CENTERING HOLES AND AS A PUNCH BEFORE DRILLING TO ENSURE AN ACCURATE DRILL START.
PUNCH, DRIVE PIN "DRIFTS"	USED FOR DRIVING OUT PINS, RIVETS, AND BOLTS. USE NEXT SIZE SMALLER THAN HOLE.

TOOL	REMARKS
REAMER, HAND, SQUARE SHANK, SPIRAL FLUTED	USED FOR ENLARGING DRILLED HOLES TO DESIRED SIZE. VARIOUS SIZES AVAILABLE.
REAMER, SQUARE SHANK, TAPER PIN, SPIRAL FLUTED	USED FOR ENLARGING DRILLED HOLES IN SHEET STOCK AND FOR REAMING SHAFT HOLES FOR TAPER PINS. IN ELECTRIC AND AIR MOTORS, REAMERS ARE EXCELLENT FOR MAKING CUTOUTS IN ALUMINUM MATERIAL.
RULE, 6-FOOT, FLEXIBLE	USED FOR SHEET METAL LAYOUT.
RULE, 6-INCH, RIGID	USED FOR SHEET METAL LAYOUT.
RULE, SQUARE, CENTER AND PRO-TRACTOR COMBI-NATION	USED FOR SHEET METAL LAYOUT.
SAW, BAND	USED FOR CUTTING FUSELAGE FRAME FORM BLOCKS.
SAW, HOLE	USED WITH PILOT IN HAND AND POWER DRILLS, THIS TOOL IS EXCELLENT FOR CUTTING CIRCULAR HOLES IN DAMAGED SHEET METAL PRIOR TO APPLICATION OF STANDARD BUTTON PATCH. SIZES RANGE FROM 5/8 TO 2 INCHES.
SCRIBE	USED FOR SHEET METAL LAYOUT.
SETS, RIVET	USED FOR BRAZIER, FLAT, AND COUNT-ERSUNK RIVET HEADS.
SKIN FASTENER, CLECO	INSERTED IN DRILLED HOLES TO HOLD TWO OR MORE PIECES OF METAL UNTIL RIVETING IS ACCOMPLISHED.
SNIPS, COMBINA-TION CIRCLE "DUCK BILL"	USED FOR STRAIGHT LINE AND CURVED CUTTING.
SNIPS, LEFT-HAND, DOUBLE-ACTION "DUTCHMANS"	USED FOR CUTTING TO LEFT AND/OR CUTTING ENDS OF TUBING.
SNIPS, TINNERS	USED FOR STRAIGHT LINE CUTTING OF SHEET METAL UP TO .064 INCH THICK.
SPOT-FACER, BACK	USED WITH VARIOUS SIZES OF PILOTS WHERE ACCURACY OF HOLE LOCATION IS DESIRED. ALSO USED FOR FLUSHING BOLT HEADS IN CASTINGS.
TAP	USED FOR THREADING VARIOUS SIZES OF HOLES FOR ALL STANDARD THREADS. AVAILABLE IN DIFFERENT DIAMETERS, THREADS PER INCH, AND PITCH.
VISE, DRILL PRESS, SPEED	USED TO HOLD OBJECTS WHILE DRILL-ING.
WHEELS, EMERY	SMOOTH AND ENLARGE INSIDE EDGES OF LARGE HOLES.
WRENCH, TAP	USED IN CONJUNCTION WITH TAPS, EZY-OUTS, HAND AND TAPERED REAMERS.

SECTION 4

MOVABLE SURFACES

1. GENERAL.

The movable surfaces consist of hydraulically operated, all-metal landing flaps and cable-controlled, fabric-covered, metal frame ailerons, elevators, and rudders.

2. LANDING FLAP CONSTRUCTION.

The centersection and outer wing landing flaps extending from left-hand aileron to right-hand aileron pass beneath the fuselage and form the lower half of the trailing edge of the wing *(see Figure 1)*. The three flap sections are constructed entirely from 24ST alclad sheet and each section consists of an .040 inch thick hat-section channel spar, an .032 inch thick Z-section leading edge, an .025 inch thick closed V-section trailing edge, and .020 inch thick ribs and skin. Where the hat-section spar is riveted to the flap skin, a torsion box is formed which reacts the main bending loads. The maximum loads on the flaps occur when the flaps are in the fully extended position. The structural part numbers of centersection and outer wing flap are illustrated *(see Figure 2)*. Each of the outer wing flaps has a length of 89-3/8 inches, a constant chord of 14-11/32 inches, and a maximum movement of 45 degrees. The centersection flap has a length of 56-5/8 inches, a constant chord of 14-11/32 inches, and a maximum movement of 45 degrees. The flaps are not statically balanced.

3. RUDDER CONSTRUCTION.

The 24ST alclad aluminum alloy rudder frame is assembled about a 1-3/4 inch diameter 24ST aluminum alloy torque tube which forms the primary load-bearing member of the structure. The component part numbers of the rudder structure are illustrated and their numbers given *(see Figure 3)*. The ribs are formed from subsequently heat-treated .020 inch thick 24SO alclad sheet. In original rib forming, power pressing is utilized to provide the integral capstrips, stiffening beads, and flanged cutouts. The ribs extend from the leading edge to the trailing edge and are riveted to the torque tube by means of 24ST aluminum alloy flanged collars. The leading edge forward of the torque tube is formed of several sheet sections of .020-inch 24ST alclad. A 3-pound cast lead counterweight attached at the top of the rudder provides adequate static balance. The trailing edge consists of a Type C144LT rolled section which is omitted for a short length to allow for the installation of a controllable trim tab. The curved top of the rudder is formed of two shallow half sections of 52S-1/2H material welded together about the common circumference and riveted to the uppermost rib of the structure. A cap, pressed from a single sheet of .032 inch thick 3SO aluminum alloy, is affixed to the lower extremity of the rudder.

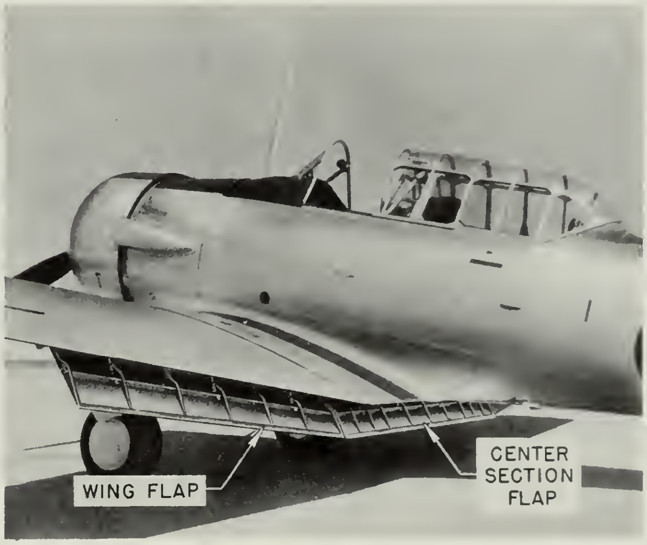


Figure 1—Landing Flaps Installed on Wing

A control horn, forged from 14ST aluminum alloy, is attached to the torque tube at the lower hinge point. Sealed-type ball-bearing hinge points are provided by means of Type B1226 eyebolts into which are pressed NA 200-K4 ball bearings. The bearings are secured in the eyebolts by annular crimping. An all-metal controllable trim tab, a trim tab operating mechanism and cable, three hinge fittings, and a control horn are assembled on the frame. The rudder frame is covered with doped fabric as outlined in a following paragraph. The complete assembly is attached to and supported by the rear spar of the vertical stabilizer *(see Figure 4)*. The rudder has an area of 13.21 square feet including 2.26 square feet of balance and .60 square feet of tab area. The rudder has an angular movement of 35 degrees to the right and 35 degrees to the left.

NORTH AMERICAN AVIATION PART NUMBERS

REF. DWGS.
- 66-18002.....FLAP - CENTER SECTION
- 55-18001.....FLAP - OUTER WING - L.H.
- 55-18001-1...FLAP - OUTER WING - R.H.

TEM	PART NO.	TITLE
1.	78-18004-1	RIB - R.H.
2.	78-18009-1	RIB - R.H.
3.	78-18009	RIB - L.H.
4.	78-18004	RIB - L.H.
5.	66-18007	EDGE - TRAILING
6.	55-18022-1	HINGE - R.H.
7.	66-18018	CHANNEL - CENTER SECTION FLAP
8.	55-18022	HINGE - L.H.
9.	66-18008	EDGE - LEADING

ITEM	PART NO.	TITLE
10.	78-18004-3	RIB - L.H.
	78-18004-2	RIB - R.H.
11.	55-18009-1	RIB - R.H.
12.	55-18004-1	RIB - R.H.
13.	55-18030	EDGE - TRAILING
14.	55-18022	HINGE - L.H.
	55-18022-1	HINGE - R.H.
15.	55-18029	CHANNEL - OUTER WING FLAP
16.	55-18031	EDGE - LEADING

Figure 2—Landing Flap Part Numbers

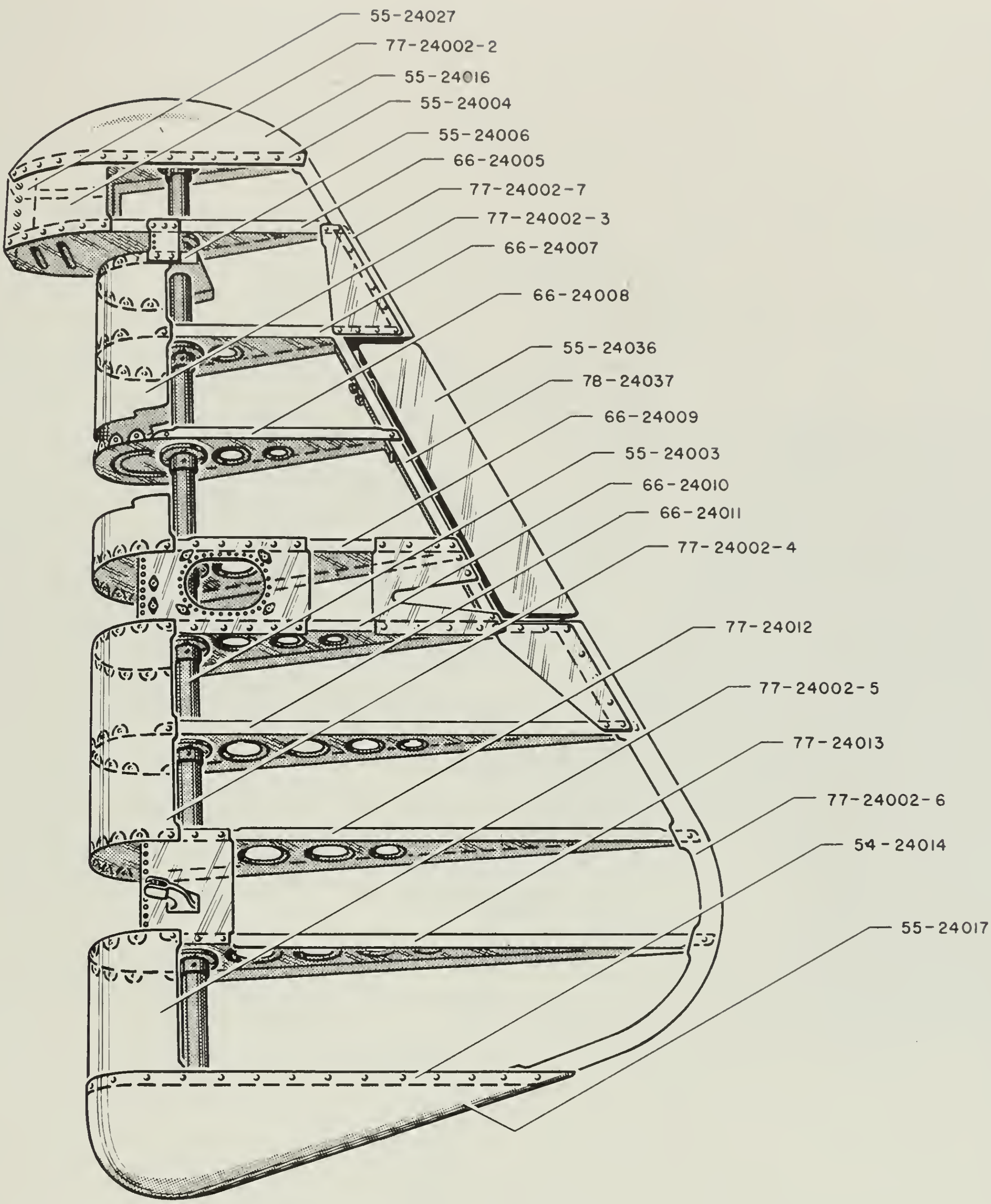


Figure 3—Rudder Frame Part Numbers

4. ELEVATOR CONSTRUCTION.

The elevator assembly consists of two interchangeable sections bolted together at the center to a short torque tube to which are attached the control horn and the center hinge points for the complete assembly. When the two sections are bolted together, the entire structure is supported by hinge brackets which are bolted to the rear spar of the horizontal stabilizer *(see Figure 6)*. The general construction and the fabric covering of each section is substantially the same as that of the rudder. The following differences from the rudder construction will be noted: The cast lead counterweight located inside the leading edge at the outboard end of each section weighs 5.25 pounds. The inboard end of each elevator frame section terminates at a rib and is therefore not provided with a cap. Type B1226 eyebolts, into which are pressed NA 200-K4 ball bearings, comprise two of the hinge points for each section. The center hinge points for the complete installation consist of sealed-type ball bearings pressed into 14ST forgings located on the interconnecting torque tube *(see Figure 6)*. The trim tab actuating cables extend through the inboard rib forward of the main torque tube of each elevator section. Each of the elevator sections has an area of 10.88 square feet including 1.75 square feet balance and .73 square foot of tab area. The entire elevator assembly

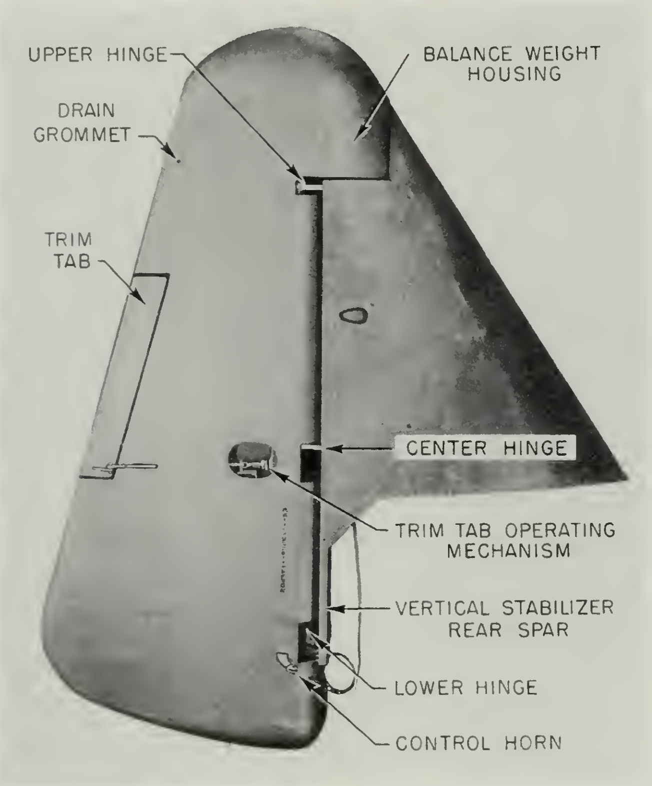


Figure 4--Rudder Installed on Vertical Stabilizer

has an angular movement of 30 degrees up and 20 degrees down. All component part numbers of the rudder structure are illustrated and their numbers given *(see Figure 5)*.

Note: If either of the elevator sections is interchanged, care must be taken to provide proper drainage for the moisture caused by condensation inside the elevator. Drain grommets are located at the trailing edge in each bay on both sides of the elevator fabric covering *(see Figure 6)*. When the elevator sections are reversed, the centers of these must be cut out on the lower side and the ones on the upper side resealed by the application of a doped fabric patch.

5. AILERON CONSTRUCTION.

The ailerons are located at the trailing edge of each wing outboard of the flaps *(see Figure 8)* and extend from wing station 147.1 to 243.4. The aileron frame structure is made entirely from 24ST alclad and is covered with doped fabric. The 24ST alclad spar and leading edge skin form a torsionally rigid box. Balancing counterweights, secured inside the leading edge skin, provide adequate static balance. A metal booster tab is located on the trailing edge to provide for ease of control. A total of three hinges supports each aileron. The flanged leading edge ribs are riveted to the leading edge skin and to the trailing edge ribs. The trailing edge ribs are formed from 24ST alclad and are provided with integral flanged capstrips in which dimpled holes are spaced to provide for the insertion of the countersunk fabric attaching screws and washers. The Grade A mercerized cotton fabric covering is attached to the ribs and at the trim tab and hinge cutouts as outlined in a subsequent paragraph. The structural parts of each aileron are illustrated and their numbers given *(see Figure 7)*. Each aileron has an area of 11.40 square feet including 2.42 square feet balance and .44 square foot tab area. Each aileron has an angular movement of 30 degrees up and 15 degrees down, utilizing a differential motion of 2 to 1.

6. RUDDER, ELEVATOR, AND AILERON ORIGINAL FABRIC COVERING.

The fabric covering of these surfaces is made from Grade A mercerized cotton fabric, doped and finished as outlined in subsequent paragraphs. The cover is machine-stitched as far as possible along the trailing edge; then the cover is slipped over the structure like an envelope and hand-sewn at the open areas as required. The fabric covering is attached to the dimpled holes in the trailing edge rib capstrips by countersunk sheet metal screws inserted through dimpled washers. Reinforcing tape is placed along the ribs under the washers and screws and finishing tape is placed over them to provide a smooth finished surface. Two metal riveting strips are used in conjunction with AN426-AD3 countersunk rivets to secure the fabric to the trim tab channel. A row of holes around the edge of each hinge cutout permits the fabric to be hand-stitched to the metal

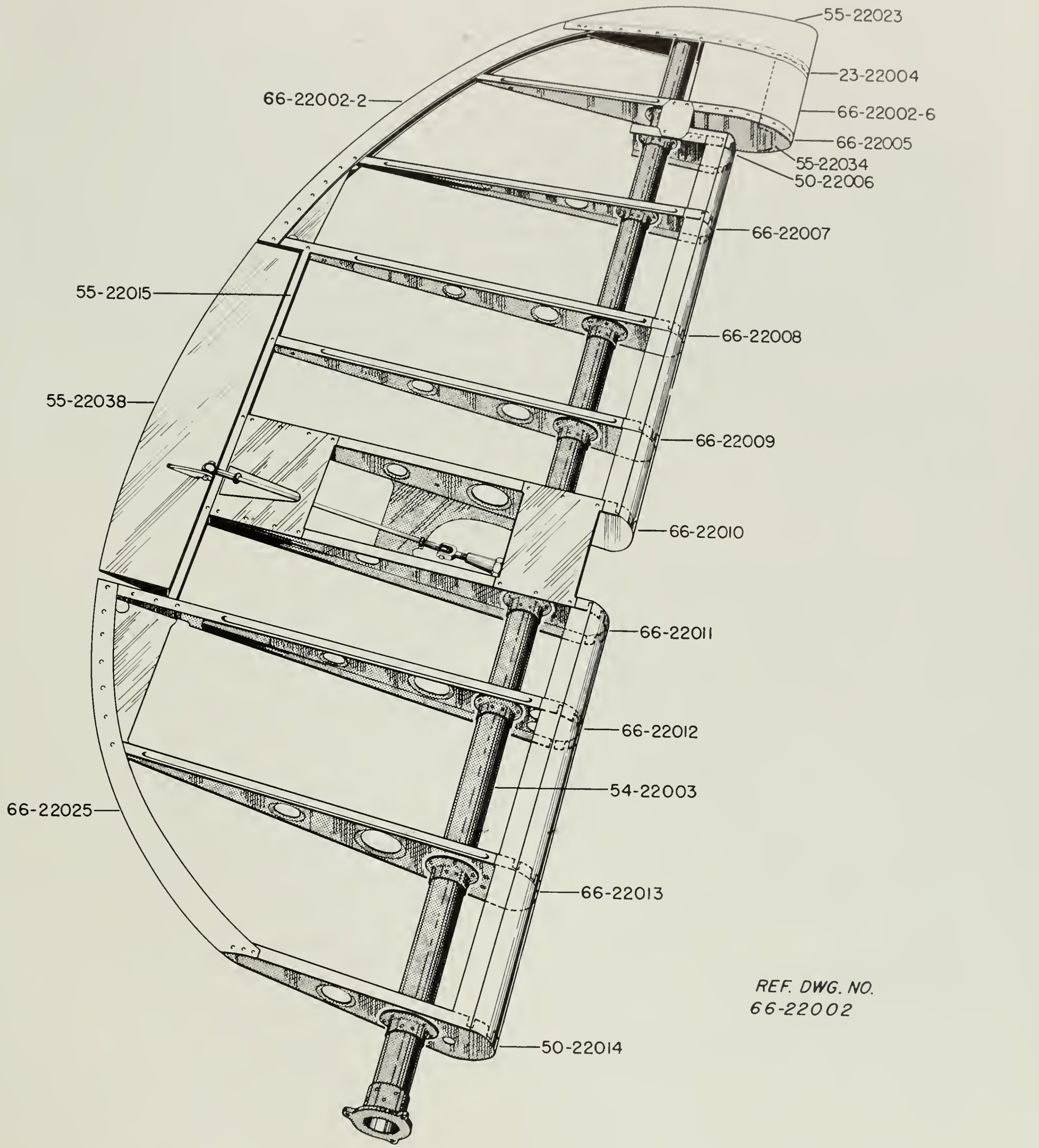


Figure 5—Elevator Frame Part Numbers

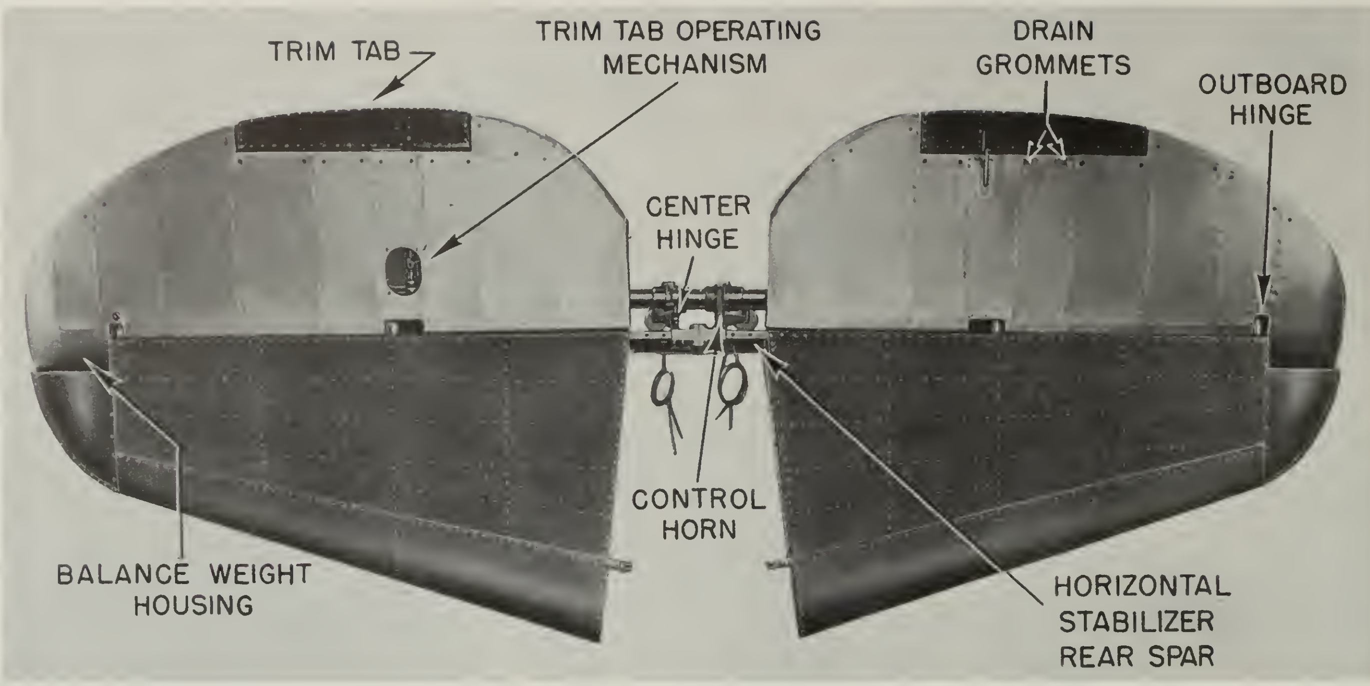


Figure 6—Elevators Installed on Complete Horizontal Stabilizer

leading edge skin at these points. Finishing tape and fabric and canvas patches are applied over all seams or rough areas to provide a smooth finished surface.

7. CONTROL SURFACE TRIM AND BOOSTER TABS.

All trim and booster tabs of the control surfaces on the earlier AT-6C Series and previous airplanes are of a riveted alclad construction, while those on the later AT-6C Airplanes are constructed of wood and phenolic fiber and are assembled with casein glue. Since neither type of these trim and booster tabs is balanced, it is very important that the tab hinges and controls have as little play as possible to prevent vibration. Severely damaged trim or booster tabs of wood or metal construction should be replaced.

8. FLAP REPAIR - GENERAL.

The flap spar, leading edge, trailing edge, and skin may be repaired as outlined in the following paragraphs. Replacement of damaged ribs is recommended; but if repair is warranted, follow the splice procedure outlined for trailing edge ribs of the wing *(see Section III)*. The landing flaps are not statically balanced in original construction and no balance check is necessary after repair. All parts of the flap are easily accessible for repair.

9. FLAP SKIN REPAIR.

Damage to the flap skin may be repaired exactly as outlined for the repair of the skin of the fixed surfaces. *(See Section III.)* If the skin directly under the flap channel spar is damaged and the damage is less than one inch in diameter, the repair may be made as illustrated *(see Figure 9)*. With a hole saw, trim out the hole. To serve as a reinforcement, cut a sheet of .032 inch thick, 24ST alclad having an approximate width and length of 1-3/4 inches. Drill out the affected rivets at each side of the damage and slip the reinforcement under the flap channel spar. With a No. 40 (.098) drill, drill the reinforcement through the existing holes in the skin and add a rivet hole between each two existing holes. Then drill two rivet holes through the reinforcement and the skin that lies between the spar channel legs. Remove the reinforcement, burr the rivet holes, and apply one coat of zinc chromate primer to all overlapping surfaces. Again insert the reinforcement beneath the channel spar and drive B1227-AD3 rivets through the channel spar leg flanges and the reinforcement plate. Insert and head an LS1127-4-4 Cherry rivet into each of the two blind holes, and trim the stems with a pair of flush-ground cutters. As previously noted, damage to other portions of the flap skin may be repaired as outlined in Section III.

10. FLAP CHANNEL SPAR.

The channel spar of the centersection and outer wing flaps may be repaired as illustrated *(see Figure 10)*. The repair outlined herein covers damage to the flap channel spar only. If the skin beneath the spar is damaged, refer to applicable paragraph for skin repairs. If the damage to the spar is extensive, cut out the damaged material and locate the cut between two skin rivets at each side of the damage. At each side of the damage, measure 4-1/2 inches and drill out the affected skin rivets in this area. To serve as splice members, prepare two sheets of 24ST alclad having a width of 2-3/4 inches and having a length equal to the damage length plus 9 inches. Bend up the splice members to a Z shape as shown, observing a minimum

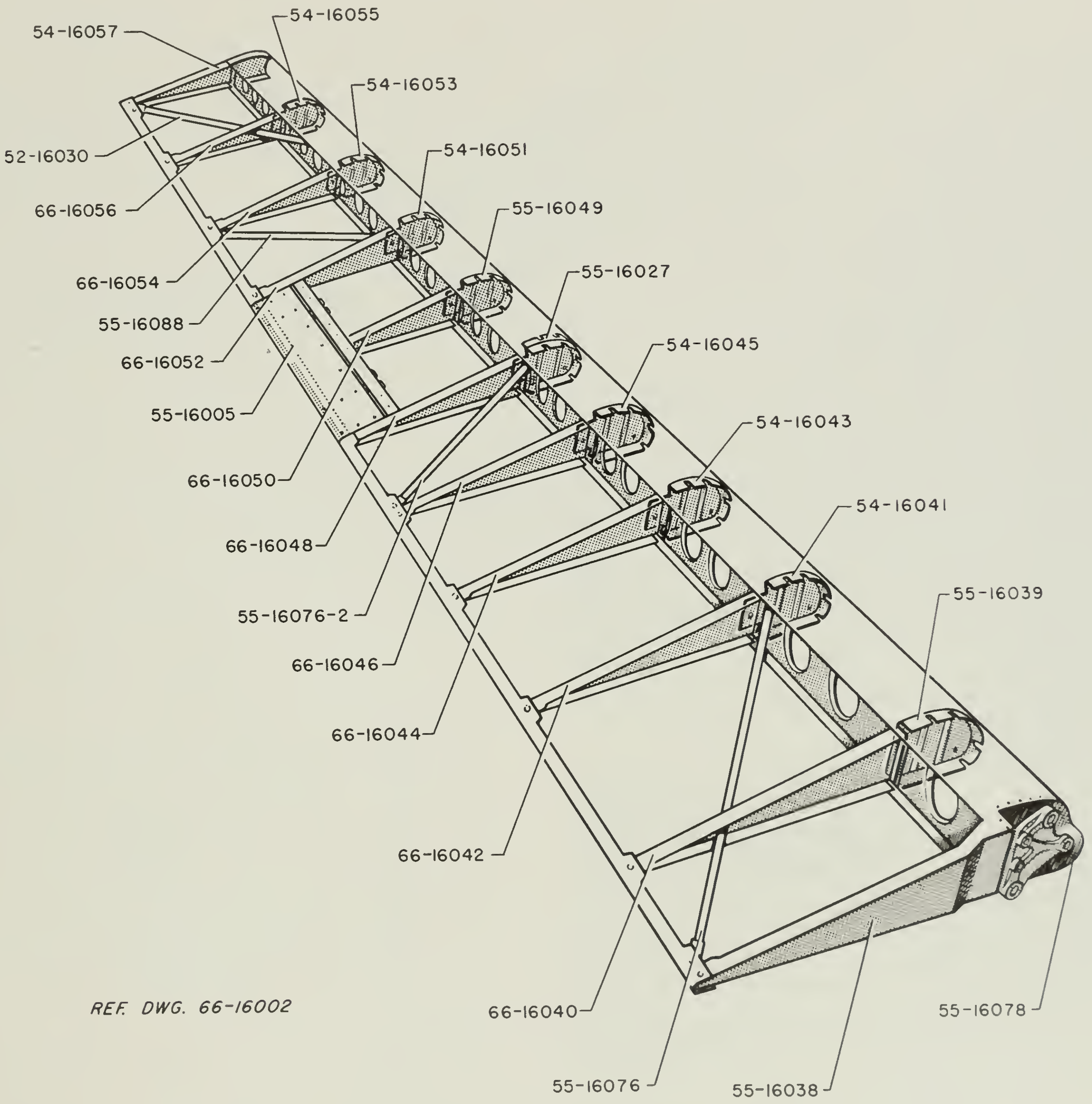


Figure 7—Aileron Frame Part Numbers

bend radius of 1/8-inch. Prepare the filler sheets as required. Apply the splice members and filler sheets to the spar, drill through several existing skin holes, and temporarily fasten the members in position. Drill the thirty-six holes through the cap and the splice members with a No. 30 (.1285) drill. Drill the splice members through the existing rivet holes in the skin with a No. 40 (.098) drill. Remove the splice members, and burr all the rivet holes. To burr the rivet holes on the inside of the hat, use a swiss needle-point file and file through the hole with a circular motion. Apply a coat of zinc chromate primer to overlapping surfaces. Resecure the splice members to the spar with skin fasteners. Replace the B1227-AD3 rivets through the skin. In the blind portion of the hat, insert thirty-six LS1127-4-4 Cherry blind rivets and pull the stems of the rivets with a G10 or G15 rivet gun. With a pair of nippers, trim the rivet stems flush with the rivet heads.

11. FLAP LEADING EDGE.

Damage to the leading edge of both the centersection and the outer wing flap may be repaired as shown *(see Figure 11)*. If the damage to the leading edge is extensive, cut out the damaged material between two skin rivets at each side of the damage. From each side of the damage, measure 3-3/4 inches and drill out the affected skin rivets in that area. To serve as a splice member, cut a sheet of .032 inch thick alclad having a width of 3-1/4 inches and a length equal to the length of the damage plus 7-1/2 inches. Bend the splice members up to the Z shape shown, observing a minimum bend radius of 1/8-inch. Prepare a filler sheet of .032 inch thick alclad having a width of 3/4-inch and a length equal to the length of the damage. Clamp the splice member and the filler to the flap leading edge, and center punch the required rivet locations at each side of the damage. Drill the twenty rivet holes through the splice member and the leading edge with a No. 30 (.1285) drill. Drill the splice member through the existing rivet holes in the skin with a No. 40 (.098) drill. Remove the splice members, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces of the splice member and filler. Reclamp the splice member to the leading edge. Through the .1285-inch diameter holes, insert and drive twenty AN442-AD4 rivets. Replace the skin rivets previously drilled out with AN426-AD3 rivets in the quantity required.

12. FLAP TRAILING EDGE.

The trailing edge of the outer wing and centersection flap may be spliced as illustrated *(see Figure 13)*. If the damage is in the form of a crack, drill a No. 40 (.098) hole at the ends of the crack. To serve as splice members, bend up two sheets of .032 inch thick 24ST alclad to the shapes shown on the diagram. Both of the splice members must be equal in length to 6 inches plus the length of the damage; one of the sheets should be 2 inches wide while the other should be 3 inches wide. Drill out the

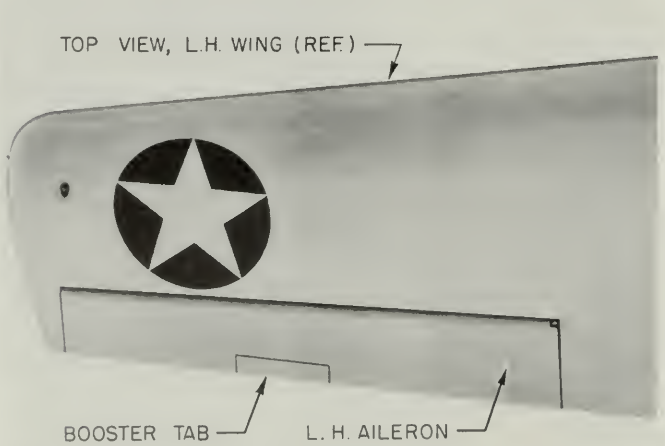


Figure 8—Left Aileron Installed on Wing

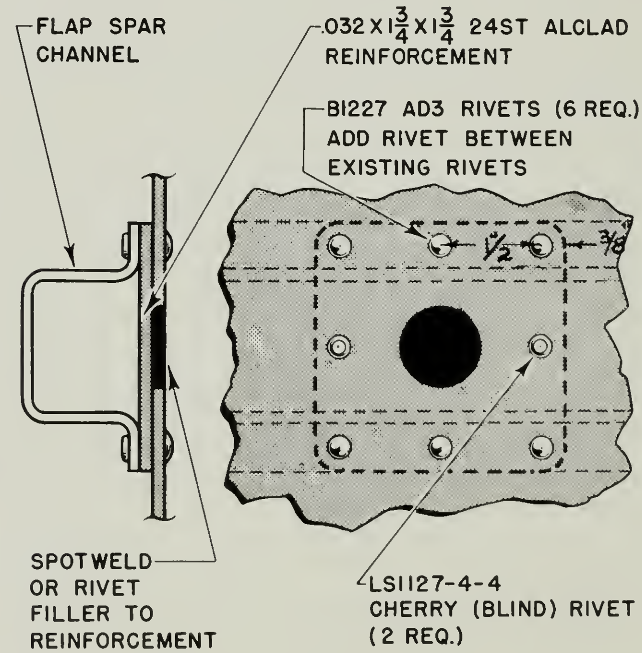


Figure 9—Repair of One-inch Diameter Hole in Skin Under Flap Channel Spar

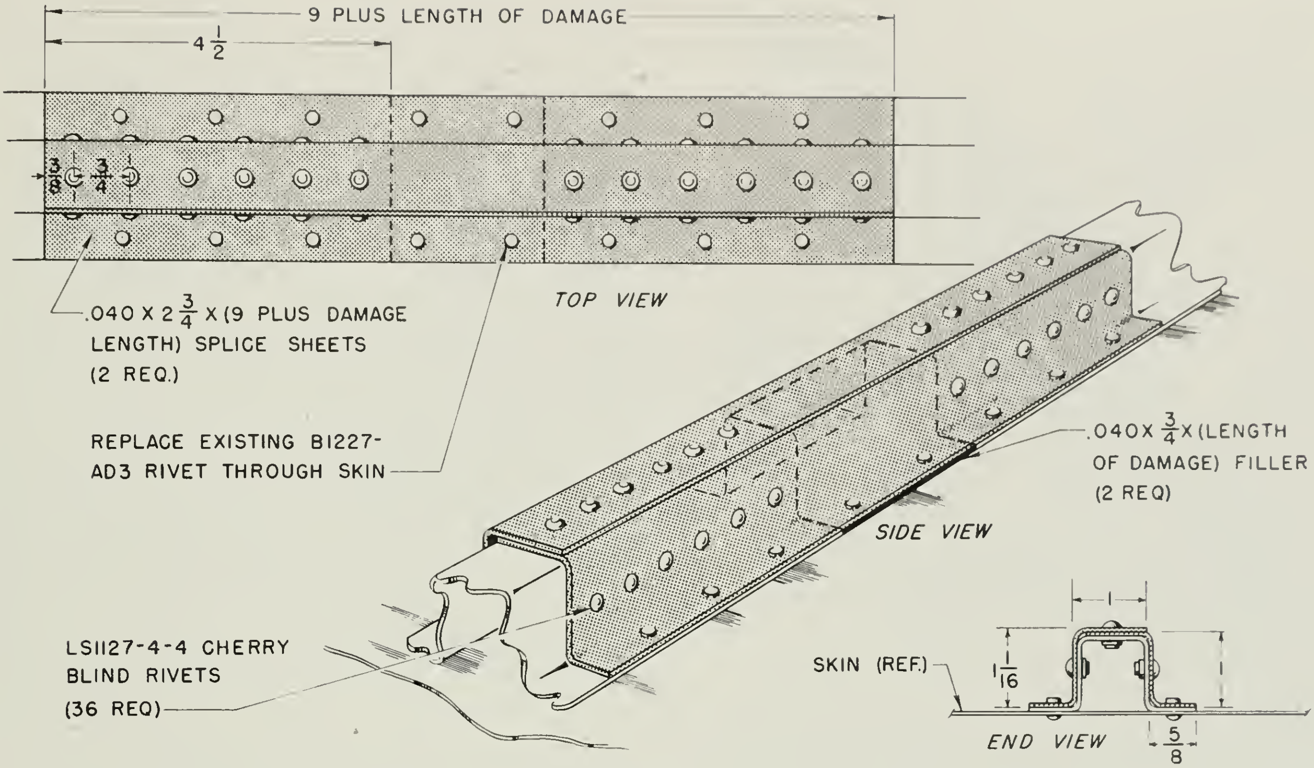


Figure 10—Flap Channel Spar Splice

affected skin rivets at each side of the damage with a No. 40 (.098) drill. Apply the splice members to the trailing edge, and drill the splice members through the existing skin rivet holes. Double the number of the skin rivet holes at the splice location by drilling an additional hole between each of the existing rivet holes. Drill the eight rivet locations through the overlap of the splice members with a No. 30 (.1285) drill. Remove the splice members, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces and temporarily secure the splice members to the trailing edge with skin fasteners. Replace the skin rivets with B1227-AD3 rivets. Through the rivet holes in the overlap of the splice members, insert LS1127-4-2 Cherry blind rivets, and expand the rivets with a G10 or a G15 Cherry rivet gun. Trim the Cherry rivet stems with a pair of nippers.

13. STRUCTURAL REPAIR OF RUDDER, ELEVATORS, AND AILERONS - GENERAL.

Because of the static unbalance created by any weight added to these surfaces, replacement rather than repair of damaged structural parts is recommended. However, where the repair material is added near the hinge line of the surface, the static balance of the surface is not seriously affected. Structural repairs to the ailerons vary from those of the elevators and rudder, as each aileron has a channel spar in place of the torque tube employed on the elevators and rudder. Obviously, access for any structural repair must be gained by cutting away the damaged fabric adjacent to the proposed repair. For fabric removal and replacement, refer to the applicable paragraphs of this Section. After a major repair, the static balance of these surfaces must be checked as outlined in a following paragraph.

14. ELEVATOR AND RUDDER RIB REMOVAL.

It is recommended that severely damaged ribs be replaced rather than repaired. Removal of an entire rib as a unit is impractical, whereas the rib may be cut and removed entirely or in part, as the damage warrants *(see Figure 12)*. To gain access to the structure, remove the covering adjacent to the damaged rib as outlined in the appropriate paragraphs of this Section. If this section of the rib aft of the torque tube is damaged, drill out the several rivets securing the end of the rib to the trailing edge

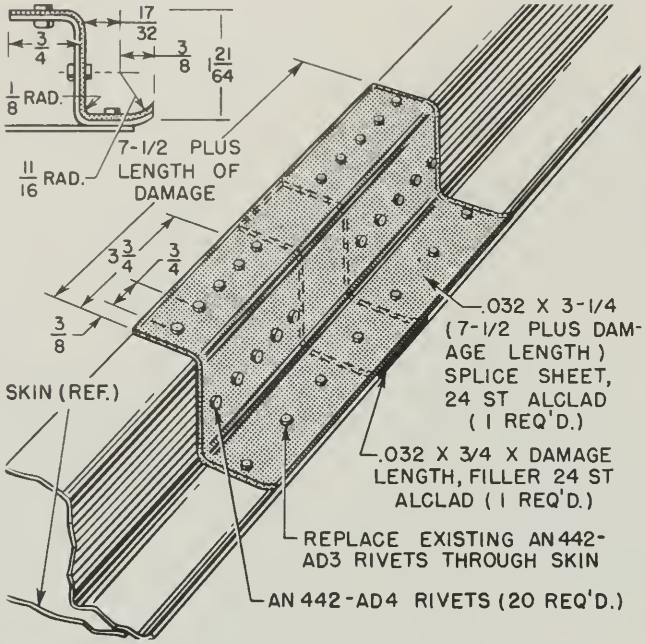


Figure 11—Flap Leading Edge Splice

strip. If the section of the rib forward of the torque tube is damaged, drill out the rivets securing the rib to the nose skin. With a hack saw, cut the rib web at the torque tube. Do not cut into the rib collar. At the collar, use a chisel to cut the remaining portion of the rib web from the edge of the flange to the torque tube. Cut carefully to prevent damage to the collar and torque tube. With a No. 40 (.098) drill, drill out the rivets securing the B1160 collar to the rib web. Lift the damaged rib section free from the structure. For the rib replacement, a North American Aviation, Inc., spare part should be utilized *(see Figures 3 and 5)*. If a spare part is not available, and immediate repair is necessary, fabricate a replacement rib section, using the damaged rib as a template. In order to prevent disturbance to the mass balance of the structure, it is essential that the weight of the new rib match the weight of the original rib.

15. ELEVATOR AND RUDDER RIB REPLACEMENT.

After the rib replacement section is prepared to match the damaged section, place the replacement section in position and drill No.40 (.098) holes through the trailing edge strip or the leading edge skin as required, using the existing holes in the structure as a guide. Secure the rib at the trailing edge or the leading edge with B1227-AD3 rivets in the quantity required. In order to splice the rib cut at the torque tube, prepare two splice sheets of .032 inch thick 24ST alclad having a maximum width of 3-1/2 inches and a maximum length of 3-1/2 inches. On each of the splice members, bend up an angle to match the existing rib caps and trim the splice members around the torque tube. Lightly center punch the rivet locations in the pattern shown *(see Figure 12)*. With a No. 30 (.1285) drill, drill the center-punched rivet locations and drill the splice sheets through the existing holes in the rib collar. Remove the splice sheets, and burr the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Secure the splice members to the rib, and insert and drive AN430-AD4 rivets through the rib collar. In the remaining holes, insert and drive AN442-AD4 rivets in the quantity required *(see Figure 12)*. Depending upon the extent of covering removed, re-cover the surface either wholly or in part as outlined in the appropriate paragraphs of this Section.

16. ELEVATOR AND RUDDER LEADING EDGE SKIN REPAIR.

After the doped fabric covering on the leading edge of the surface has been properly removed from the damaged area, small holes in the

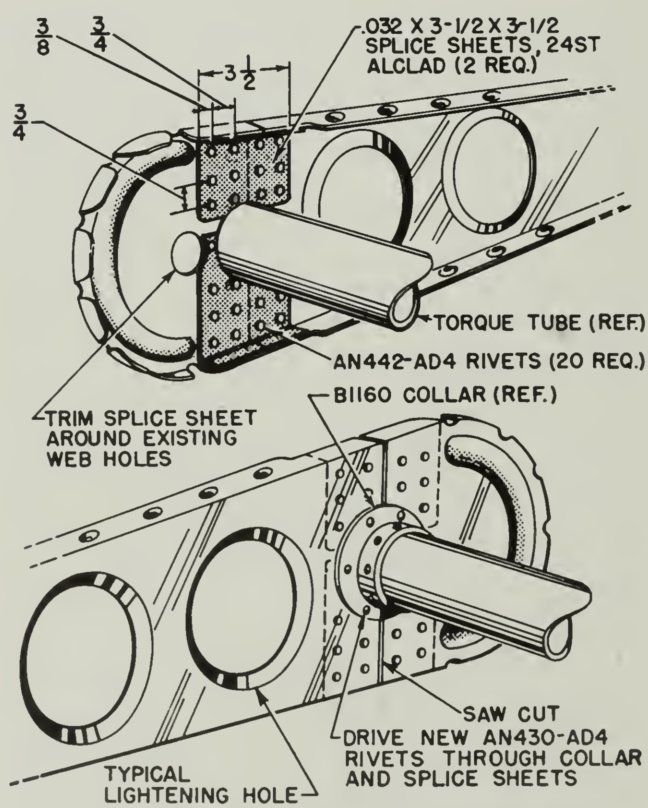


Figure 12—Rudder and Elevator Rib Replacement

leading edge skin may be repaired as set forth in the applicable paragraphs in Section III. If the damage is extensive, remove the entire affected leading edge section by drilling out the rivets securing the skin to the ribs. Roll an equivalent leading edge skin section from .020 inch thick 24ST alclad sheet. Apply the replacement skin to the proper position on the leading edge ribs; and with a curved scribe, mark the rivet locations on the inside of the skin. Remove the skin, and drill the skin rivet locations with a No. 40 (.098) drill. Apply one coat of zinc chromate primer to all overlapping surfaces of the replacement skin. Again apply the replacement skin to the proper position on the leading edge ribs and fasten in place with skin fasteners. Through the rivet holes in the skin replacement section, insert and drive AN426-AD3 or B1227-AD3 rivets as originally used through the skin. Rivet first at the absolute leading edge and then around the periphery of the skin. Remove the skin fasteners as the drilling progresses. Apply fabric covering to the exposed structure as outlined in the applicable paragraphs of this Section.

17. RUDDER, ELEVATOR, AND AILERON TRAILING EDGE STRIP REPLACEMENT.

As all additional weight in the form of repairs should be avoided at the trailing edge of a structure, replacement rather than repair of trailing edge strips is recommended. After the doped fabric covering has been removed as set forth in the corresponding paragraphs of this Section, drill out all rivets that secure the trailing edge strip to the ribs and to any longitudinal stiffening tubes. A North American Aviation, Inc., spare part should be used for replacing the damaged strip. *(See Figures 3, 5, and 7.)* However, if none is available, fabricate a replacement strip of 24SO alclad. Bend up a V-formed section to match the length, thickness, weight, and cross section of the removed damaged strip. In order to observe the minimum bend radius allowable, use 24SO alclad and subsequently heat treat to required hardness. *(See Section I.)* Place the strip in position; and with a blind hole locating tool *(see Section III)*, mark the locations of the holes for the rib attaching rivets. Remove the strip, drill and burr the holes, and apply a coat of zinc chromate primer to all faying surfaces. After replacing the strip in position, insert two or more skin fasteners to hold it rigidly in place. Insert and upset all attaching rivets. With Grade A mercerized cotton fabric, re-cover the partially

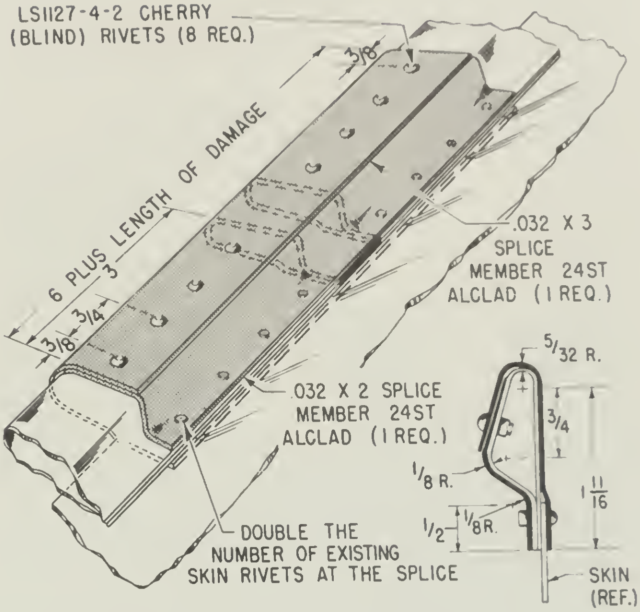


Figure 13—Flap Trailing Edge Splice

exposed structure as set forth for partial fabric re-covering in a following paragraph.

18. AILERON SPAR SPLICE.

If the aileron spar is damaged, it cannot be replaced satisfactorily, but the spar can be repaired without seriously affecting the mass balance of the aileron as a whole. It is obvious that the entire fabric covering or a section of the covering must be removed in order to gain access to the spar for repair. For the removal of covering, refer to the appropriate paragraph of this Section. After the covering has been removed, examine the damage to the spar. If the damage is not too extensive, cut out the damaged material between two skin rivets at each side of the damage. On both the upper and the lower cap of the spar, measure 3-3/4 inches on each side of the damage, and drill out the affected skin rivets securing the cap in this area *(see Figure 14)*. To serve as splice members, two sheets of .032 inch thick 24ST alclad are required, each having a maximum width of 3 inches and a maximum length of 7-1/2 inches plus the length of the damage. Along the length of one of the splice members, bend up an 80-degree, 3/4-inch flange. Along the length of the other splice member, bend up an 87-degree, 3/4-inch flange. Observe a minimum bend radius of 1/8-inch. Trim the splice members to fit the spar, and trim around the splice members to match any existing lightening holes in the affected area. Clamp the splice members to the spar and center punch the required twenty rivet locations at each side of the damage. Center punch other rivet locations to suit the spar depth. Drill the center-punched rivet locations with a No. 30 (.1285) drill. Drill the splice members through the existing skin rivet holes. Remove the splice members, and burr all the rivet holes. Apply one coat of zinc chromate primer to all overlapping surfaces. Reclamp the splice members to the spar; and through the skin rivet holes, insert and drive AN426-AD4 rivets in the quantity required. *(See Figure 14.)* Through the remaining holes, insert and drive AN442-AD4 rivets. Depending upon the extent of covering removed, re-cover the surface either wholly or in part as outlined in the appropriate paragraph of this Section.

19. AILERON TRAILING EDGE RIB REPLACEMENT.

The simplicity of the construction of the aileron trailing edge ribs and the ease of their attachment to the aileron spar warrants replacement rather than repair of damaged ribs. Rib replacement maintains the static balance of the aileron, as no resultant weight from repair members is added to the trailing edge portion of the structure. Most of the aileron trailing edge ribs are secured to the aileron trailing edge by two B1227-AD3 rivets, to the aileron spar web and to the extending flange on the leading edge ribs by four AN442-AD3 rivets, and to the spar flange and overlapping leading edge skin by two AN426-AD3 rivets. Obviously, if the trailing edge rib is damaged enough to warrant replacement, the surrounding fabric will also be damaged and must be removed and repaired as outlined in the applicable paragraphs of this Section. Remove all fabric necessary to obtain access to the damaged rib area. Drill out the eight rivets attaching the rib to the structure. If a stiffening tube extends through the damaged ribs, drill out the attaching rivets on one end of the tube to permit the damaged rib to be lifted free and the new rib to be inserted into position. When removing any of the three ribs which support the booster tab supporting channel, drill out the two additional rivets securing this channel to the rib. For replacement of the damaged rib, a North American Aviation, Inc., spare part should be utilized. *(See Figure 7.)* However, if a spare part is not available, fabricate a replacement rib from a sheet of 24SO alclad of the required thickness and heat treat to required hardness after

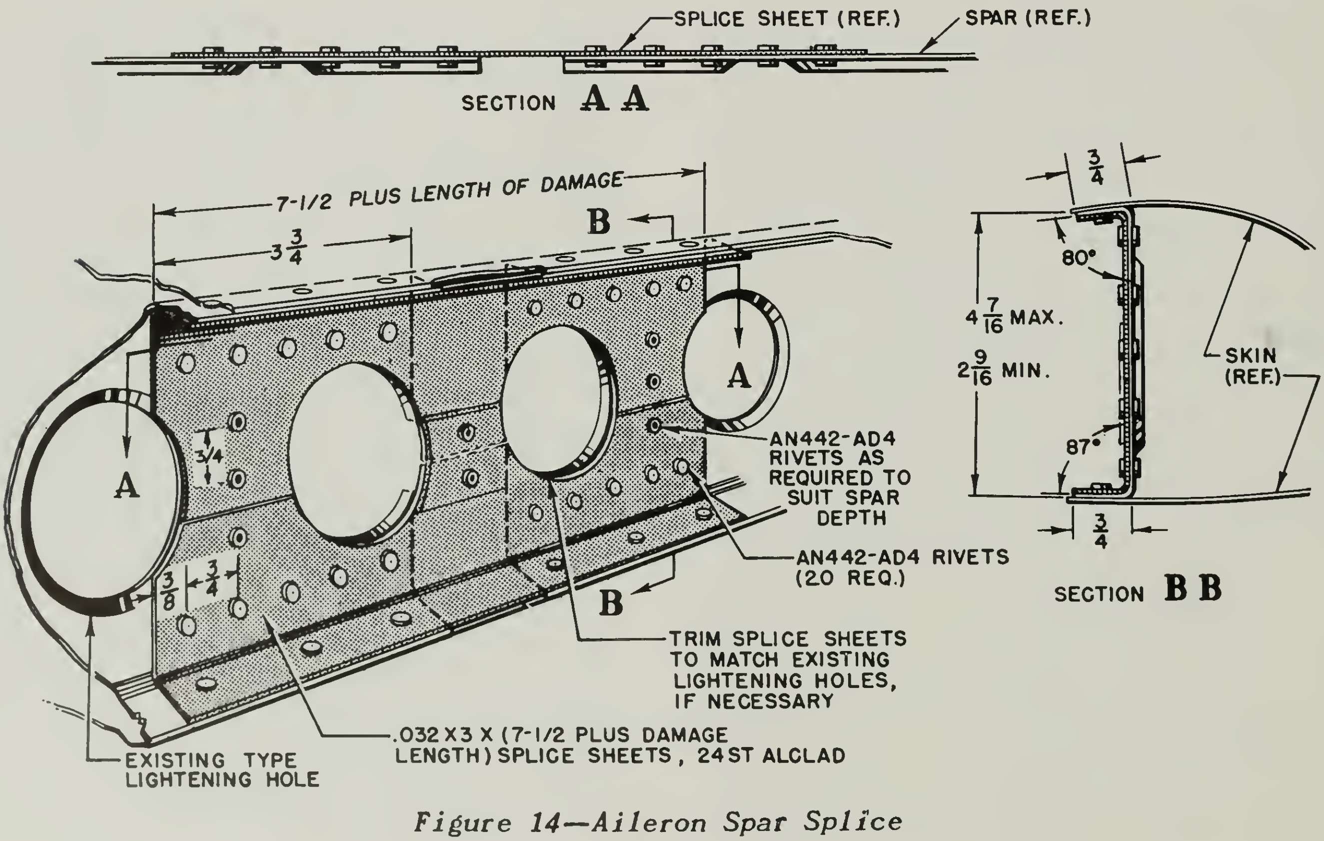


Figure 14—Aileron Spar Splice

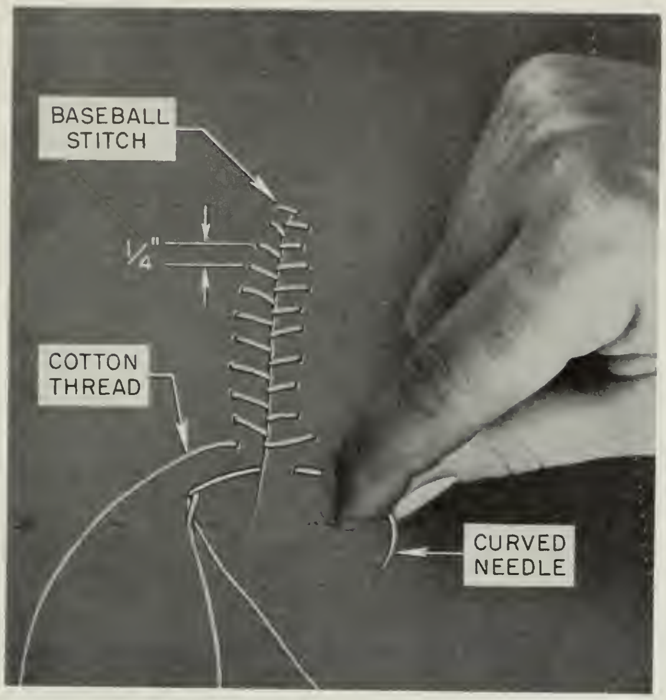


Figure 15—Sewing Small Fabric Tear

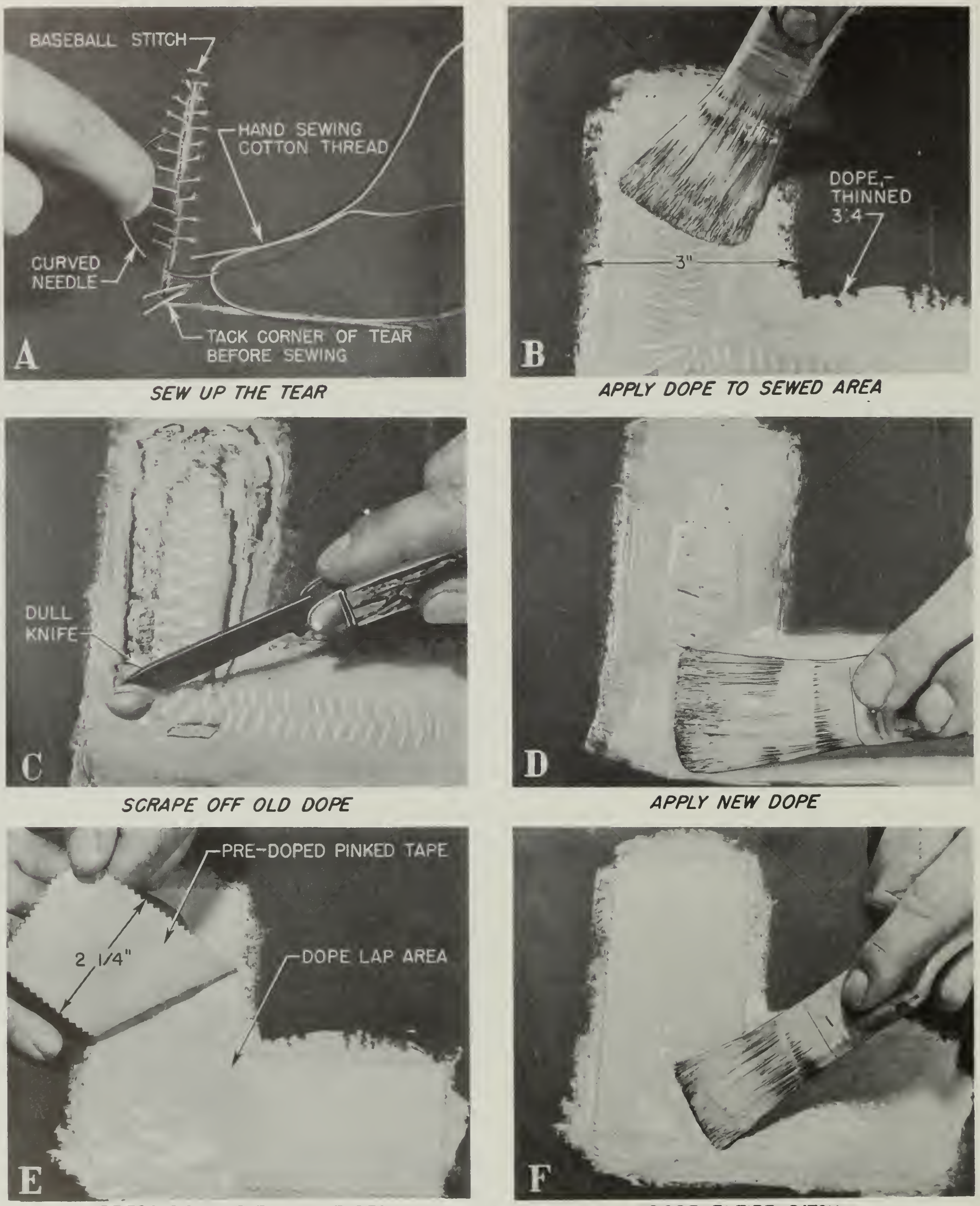
forming. *(See Section I.)* Observe a minimum bend radius of 1/16-inch and round all corners to a 1/4-inch minimum radius. After the replacement rib has been obtained or formed, insert it into position; and wherever possible, mark and drill all rivet holes through the existing holes in the structure. Use a curved scribe or a blind hole locating tool *(see Section III)* for marking the hole locations of the two rivet holes in the spar web. After marking these hole locations, remove the rib and drill these two holes. Burr all rivet holes and apply one coat of zinc chromate primer to all faying surfaces of the replacement rib. Again place the rib in position. Insert skin fasteners in the rivet holes to hold the rib rigidly in place; then insert and drive all attaching rivets, removing the skin fasteners as the drilling progresses. By the use of special bucking bars, manipulated through the lightening holes in the spar, buck the rivets securing the rib to the web of the aileron spar. After the rib has been riveted in position, re-cover with fabric the exposed area of the structure by following the directions set forth for fabric repairs in the appropriate paragraphs of this Section.

20. AILERON NOSE RIB REPLACEMENT.

In order to remove and replace a damaged aileron nose rib, it will first be necessary to remove the leading edge fabric and skin from the damaged rib area. To gain access to the

Figure 16—Cutting Out Damaged Fabric

interior of the leading edge skin through the lightening holes in the aileron spar, remove enough fabric from the adjacent trailing edge bay to permit proper tool manipulation. With a hack saw or a tapered reamer, cut away the damaged skin as set forth in a following paragraph. Drill out the attaching rivets of the nose rib and remove the rib from the structure. A North American Aviation, Inc., spare part should be used for replacing the damaged rib *(see Figure 7)*. However, if such a spare part is not available, fabricate a replacement rib from 24SO alclad sheet of the required thickness and heat treat to the required hardness after forming *(see Section I)*. Observe a minimum bend radius of 1/16-inch and round all corners to 1/4-inch minimum radius. After the replacement rib has been obtained or formed, insert it into position; and wherever possible, mark and drill all rivet holes through the existing holes in the trailing edge rib and nose skin. By the use of special bucking bars *(see Section I)*, manipulated through the lightening holes in the spar, buck the rivets securing the rib to the leading edge skin. After the rib has been securely riveted in position, repair the leading edge skin and replace the removed



SEW UP THE TEAR

APPLY DOPE TO SEWED AREA

SCRAPE OFF OLD DOPE

APPLY NEW DOPE

PRESS DOWN PATCHING TAPES

DOPE ENTIRE PATCH

Figure 17—Patching V-shaped Fabric Tear

fabric by following the instructions set forth in the applicable paragraphs of this Section.

21. AILERON LEADING EDGE SKIN REPAIR.

After the doped fabric covering on the leading edge of the surface has been properly removed from the damaged area, small holes in the leading edge skin may be repaired as set forth in the appropriate paragraphs in Section III. However, if the damage is extensive, the leading edge skin of the aileron may be repaired by removing the entire damaged skin between the two adjacent ribs and by splicing in a new section of skin cut and formed from 24ST alclad sheet. To gain access to the damaged skin area and to provide for proper tool manipulation, first remove the doped fabric covering from the leading edge and adjacent trailing edge bay or bays by following the directions set forth in the applicable paragraphs, in this Section, on removing fabric. With a hack saw, metal snips, or a tapered reamer, cut away the damaged skin within 7/8-inch of an adjacent rib flange rivet line. Drill out the affected skin rivets around the circumference and also drill out the rivets that attach the skin to the spar flanges. To serve as a splice section for the leading edge skin, cut and form a sheet of .025 inch thick 24ST alclad to match the width of the removed damaged portion. The length of the splice section should be 2-1/2 inches longer than the removed portion to permit the ends to overlap the rib flanges at least 1-1/4 inches, permitting the new skin to be lap riveted to the rib flanges. After the splice member has been cut and formed to fit the contour of the leading edge, lay it in position over the nose ribs, and with a curved scribe, mark all rivet hole locations through the existing holes in the ribs and skin. At each side of the damage, mark an additional rivet line on the lap area adjacent to the rib flange. Along this rivet line, center punch and drill rivet locations at one inch on centers. After the holes are thus marked, remove the skin splice member and drill all existing rivet holes. Burr the rivet holes and apply one coat of zinc chromate primer to all overlapping surfaces. Reapply the splice member and secure it in place by the use of skin fasteners in alternate rivet holes. Insert and drive all attaching rivets, starting at the tip of the leading edge to prevent the skin from buckling as it is riveted around the contour of the leading edge. After riveting is accomplished, re-cover the exposed structure with fabric by following the directions set forth in the applicable paragraphs in this Section.

Figure 18--Damage Requiring Fabric Section Replacement

22. GENERAL FABRIC PATCHING.

Minor patching of small tears, V-shaped tears, and small holes may be accomplished in the conventional manner. Such repairs, and all fabric re-covering procedures from replacement of one section to re-covering of the entire control surface, are set forth in the following paragraphs. When repairing tears and holes or replacing entire sections, the repairs must develop the full strength of the surrounding material. Patches should match the adjacent areas in appearance. This may be accomplished by applying the same number of coats of dope to the patch as were applied to the surface originally. Sand lightly with No. 7/0 sandpaper between each coat. It is to be noted that care should be taken when applying additional coats of dope or finish to the complete surface, as such applications add weight to the surface, causing disturbance to the static balance. The amount of material added in the form of patches is limited by the resultant fabric surface tension and by the static balance limitations. Patches must never overlap another patch. If a large patch is not feasible or if the patch area must exceed half the bay section area, the complete affected fabric section must be replaced.

23. TESTING FABRIC FLEXIBILITY.

The necessity for re-covering fabric-covered

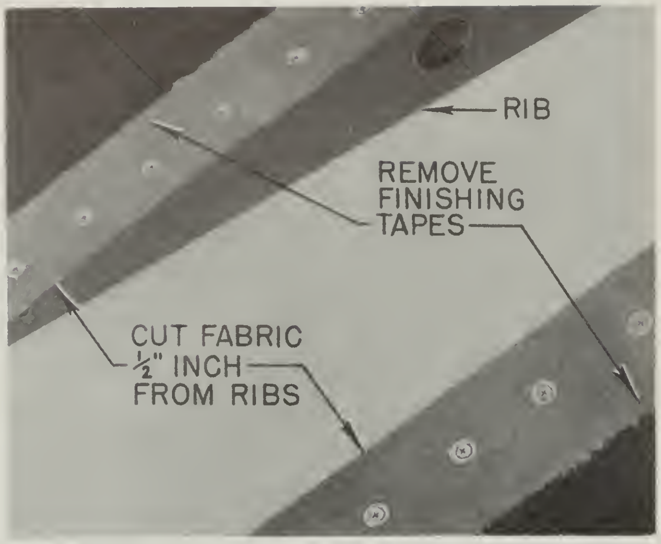


Figure 19—Damaged Fabric Section Removed

control surfaces is made evident by cracked finish, loss of tautness and flexibility, excessive patching, extensive damage, insecurity of attachment, etc. The flexibility of the material is an accurate index to its service expectancy. At regular intervals, therefore, the flexibility of fabric covering should be tested. This may readily be accomplished by pressing heavily an open area with the thumb. If the dent thus created disappears within a few minutes, the fabric is quite new. If the dent requires an hour or more to disappear, service expectancy of the fabric is short. If the fabric cracks where it has been dented, the finish is dead and will require re-covering. Rejuvenation in this case, or in any other case, is not recommended and should not be attempted. Also, partial re-covering will be required where the summed-up areas of existing patches between ribs exceed 100 square inches.

24. SMALL FABRIC TEARS.

A satisfactory repair for small straight fabric tears may be accomplished by sewing the torn edges tightly together, using a baseball stitch. Sewing the tear prevents further ripping of the fabric during flight and should not be omitted unless the tear is less than one inch in length. Merely applying a doped fabric patch over the tear is not sufficient and will result in a loss of tautness of the fabric covering, especially when two or more tears appear in one section. Use a single strand of No. 8, 4-ply hand-sewing cotton thread and a curved needle. Knot the end of the thread and start sewing at one end of the tear. Using a baseball stitch, space the stitches not more than 1/4-inch apart and stitch 1/4-inch back into the cover away from the edge of the tear *(see Figure 15)*. As the seam is made, draw the edges tightly together but do not pull the thread too taut or ripping of the fabric will result. After the stitching is completed, proceed with cleaning, doping, and patching as outlined in the following paragraph for V-shaped fabric tears. One strip of pinked finishing tape of the required length adequately serves as a patch for this type of tear. This strip of finishing tape should extend at least 1-1/4 inches beyond each end of the tear.

25. V-SHAPED FABRIC TEARS.

If the opening of the tear is V-shaped, as is often the case, the tear may be patched by sewing the edges of the tear together and applying doped pinked tape over the stitches. Tack stitch the torn corner of the tear before starting the sewing so that the edges of the fabric will be held in place while the sewing is being accomplished. Use a single strand of No. 8, 4-ply hand-sewing cotton thread and a curved needle. Knot the end of the thread and start the stitching so the knot will be left on the inside of the fabric. Start sewing at one end of the tear and continue through the tack stitching at the corner until the complete tear has been sewed. Using a baseball stitch, space the stitches not more than 1/4-inch apart and stitch 1/4-inch back into the cover away from the edge of the tear *(see Figure 17, Detail A)*. As the seam is made, draw the edges of the tear tightly together to maintain the original tautness of the fabric section. Do not pull the thread too taut or ripping of the fabric will result, especially if the fabric has been in service and thus exposed to the elements. After the tear

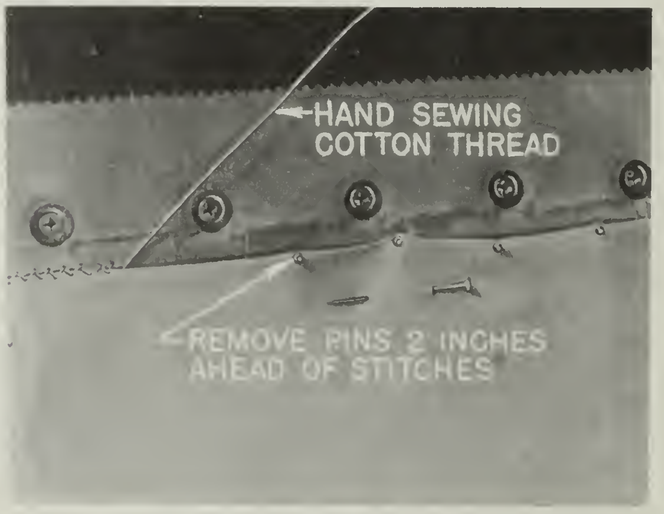


Figure 20--Final Folding, Pinning, and Sewing of Fabric Section

has been satisfactorily sewed, clean the surface over which the repair tape will be applied. Remove any dirt, grease, or paint by rubbing the surface with a rag dipped in dope, dope solvent, or acetone. Wipe the surface with a clean rag. Care must be taken to prevent dope solvent, thinned dope, or acetone from dripping through to the inside of the opposite surface, thus causing the dope to blister. To avoid this, tip the surface straight up, or turn the unit over and apply from the lower side. To remove satisfactorily the clean and semipigmented dope of the original finish, apply a wet coat of thinned dope to the patch area *(see Figure 17, Detail B)*. Allow the thinned dope to soak into the fabric for 10 minutes. After the softening of the old dope of the original surface has taken place, scrape off all applied and original dope with a dull knife *(see Figure 17, Detail C)*. Cut a patch of predoped pinked tape or fabric of sufficient size to cover the tear to a point at least one inch beyond the tear in all directions. Predoped 2-1/4 inch pinked finishing tape will often serve satisfactorily for patching V-shaped tears. Apply one brush coat of clear dope to the cleaned surface and press the predoped patch into position *(see Figure 17, Details D and E)*. Make certain that all patch edges are firmly stuck down. Apply one brush coat of dope to the entire patch *(see Figure 17, Detail F)*. After this coat of dope is dry, proceed with the application of the second brush coat of clear dope, the spray coat of clear dope, and the two spray coats of semipigmented aluminized dope as directed in a subsequent paragraph.

26. LARGE TEARS AND HOLES IN FABRIC.

Large tears and holes in fabric coverings may be repaired by removing the damaged fabric and sewing in a patch. In all cases of badly frayed tears, large tears, or large holes, cut out the damaged portion of the fabric to a triangular- or rectangular-shaped opening *(see Figure 16)*. Better seams, binding the new fabric to the old, can be made along the straight sides of the resultant opening. After the fabric has been cut out as directed, measure the opening and cut a piece of new fabric to extend at least 1/2-inch beyond the opening on all sides. Fold the edges of the new fabric under 1/2-inch and sew to the edges of the opening. Temporary stitches at each corner facilitate seam stitching. Use a single strand of No. 8, 4-ply cotton hand-sewing thread and space the stitches close together. Use an overcast or a herringbone stitch. The tautness of the patch fabric should be such that, after doping, the surface tension will equal that of the original surface. Proceed with sewing, cleaning, tape application, and doping as outlined in the preceding paragraph. No more than one large hole should be repaired in any one section.

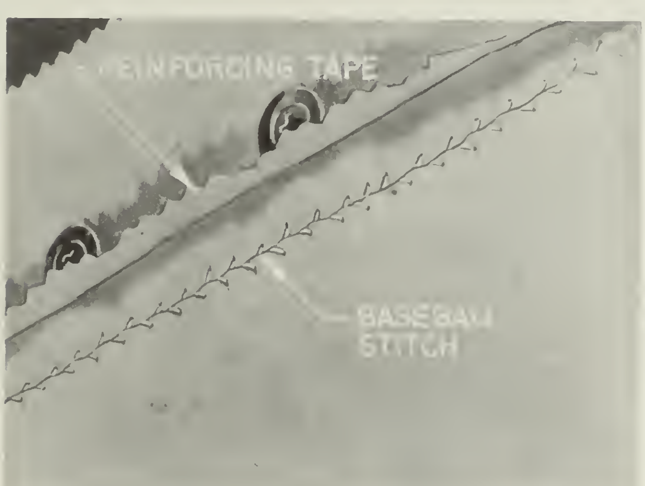


Figure 21—Detail of Fabric Section Stitching

Figure 22—Rubbing Down Reinforcing Tape at Fabric Section Seams

27. FABRIC SECTION REPLACEMENT.

Occasions may arise when an entire fabric bay section between ribs, from leading edge to trailing edge, must be replaced because patches overlap or exceed one-half the bay section area, resulting in static unbalance or loss of surface tautness. Fabric sections should also be replaced if the damage is large and the patch area would exceed 100 square inches *(see Figure 18)*. To remove and replace a damaged fabric section, proceed as follows: Remove the finishing tape from the ribs on both sides of

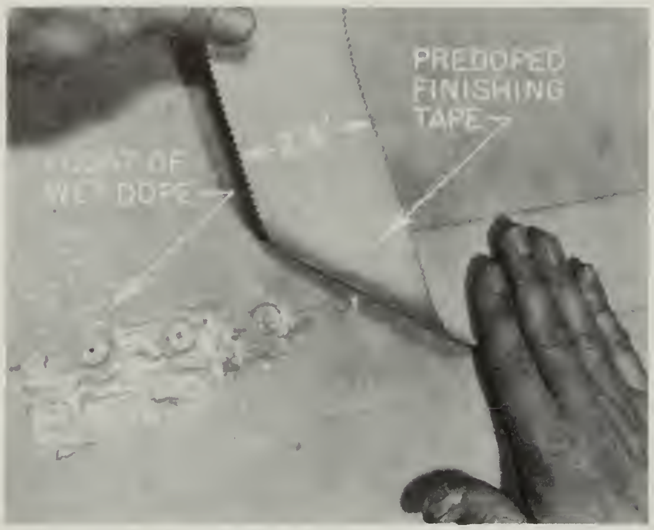


Figure 23--Applying Finishing Tape on Fabric Section Seams

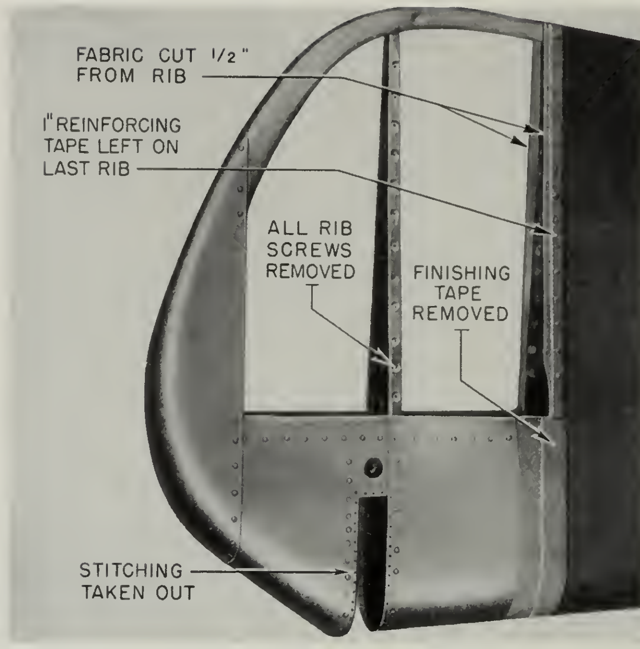


Figure 24--Typical Removal of Fabric From Two or More Sections

the damage by pulling away from the damaged area. Pulling toward the damaged area will result in damage to adjacent sections. Remove the trailing edge tape in a similar manner. Draw a line 1/2-inch in from each adjacent rib and cut out the damaged section along these lines from the leading edge to the trailing edge *(see Figure 19)*. Pry up and turn back the reinforcing rib tape. Cut a piece of new fabric which will extend from a point 2 inches beyond the centerline of the leading edge to a point one inch beyond the centerline of the trailing edge. The fabric must be wide enough on the leading edge section to match the pinked edges of the finishing tape for that section. Allow for a 1/2-inch fabric turn-under for sewing at the rib section. Apply a brush coat of dope to that part of the nose section to be re-covered; align the new fabric and press into position. Allow to dry. Pull the rib section of the fabric taut and pin in position. Space the pins evenly around the edge of the section so as to maintain a uniform tightness. Starting at the spar, or just aft of the leading edge skin, turn under the remaining strip of original fabric along the rib; turn under the edge of the new fabric as mentioned above and start sewing at the edge of the leading edge skin. This will be just aft of the spar in the case of the aileron. Use a herringbone or a baseball stitch and sew toward the trailing edge, removing pins as the sewing progresses *(see Figure 20)*. The seam must be close enough to the rib to come under the edge of the rib finishing tape or, more preferably, under the edge of the reinforcing tape *(see Figure 21)*. Follow the same process for the opposite rib. Remove the pins along the trailing edge; apply a coat of dope to the trailing edge and stick down the remaining edge of the section, allowing the fabric to extend beyond the centerline of the trailing edge at least one inch. After all edges of the new fabric section have been either sewed or doped in place, apply one brush coat of dope over the stitching along the ribs and rub down the reinforcing tape *(see Figure 22)*. Allow to dry. Apply a brush coat of dope over the entire new panel; and while the dope is still wet, lay the finishing tape on the ribs and on the leading and trailing edges *(see Figure 23)*. Apply drain grommets if necessary. Follow routine doping and painting procedure set forth in a subsequent paragraph. Allow to dry and cut out the centers of all drain grommets.

28. PARTIAL FABRIC RE-COVERING.

In the event that two or more adjacent fabric bay sections are damaged, or if the damaging projectile or object has gone through both sides of the fabric covering, the static balance of the control surface may be more easily retained by removing the entire damaged portion of the old fabric surface and by re-covering the exposed structure. When re-covering, utilize the presewed, partial envelope method. Cutting the old fabric may be accomplished by two different methods, and the method used will be dependent upon the location and nature of the damage. When partial re-covering is practical and when the damage is near a rib, remove the fabric from both sides of all sections that are to be re-covered *(see Figure 24)*. Remove the finishing tape from the area concerned, taking care to pull towards the undamaged portion of the surface when removing the tape adjacent to the remaining portion. This prevents damage to the adjacent fabric sections. After the finishing tape is removed, screw out all fabric attaching rib screws *(see Figure 25)*. With a sharp knife, cut the stitching around any hinge or access cutouts present in the sections. Do not remove the screws and reinforcing tape from the rib enclosing the adjacent undamaged section of the fabric covering. With a sharp knife or a pair of scissors, cut the fabric to within 1/2-inch from this rib *(see Figure 24)*. Fold this fabric under in a straight line along the rib. Fabric attachment along this rib will follow the method set forth and illustrated in the preceding paragraph. Prepare a partial envelope cover from new fabric large enough to cover the stripped area of the structure and long enough to lap over the undamaged area at least 2 inches. This extra fabric is needed for pinning, folding, and sewing operations. Slip the partial envelope cover over the structure and pull taut *(see Figure 26)*. Pin the edges in place and proceed with the hand sewing as set forth and illustrated in the corresponding paragraphs in this Section. Attach the reinforcing tape to all ribs under the new fabric, using the screws and dimpled washers that were previously removed. Apply all finishing tape, canvas and fabric patches with the second coat of dope. For the proper location and dimensions of tape, patches, and drain grommets, refer to the corresponding fabric re-covering diagram *(see Figures 27, 28, and 29)*. Follow routine doping and painting procedure set forth in a subsequent paragraph. If the damage is not near a rib, the fabric may be cut further from the rib and the upper and lower fabrics sewed together *(see Figure 25)*. Pinning, sewing, and doping procedures are the same as previously outlined.

29. REMOVAL OF ENTIRE FABRIC COVERING.

To remove the old fabric covering from the entire structure, first tear the trailing edge tape free. Then lift the rib finishing tape and tear it free. Remove all other finishing tape around cutouts and over seams. Below the rib finishing tape are located Phillips head, countersunk, self-tapping screws and dimpled washers sunk into recesses in the rib capstrip. Extract these screws by unscrewing them with a No. 1 Phillips screwdriver. Care must be taken to screw out and not pull out all rib screws. Pulling the screws out enlarges the holes in the rib, and a larger screw must be used when replacing the fabric. These larger screws add weight aft of the hinge line and their use should therefore be avoided. Pass a sharp knife about the trailing edge to cut all machine and hand stitching. Drill out the AD3 countersunk rivets in the two metal riveting strips at the trim tab cutout. Peel the covering carefully forward, cutting attaching threads about the circumference of access holes and hinge cutouts as necessary. After the covering is removed, clean the frame thoroughly with dope thinner and a clean cloth. Inspect the entire structure carefully for damage.

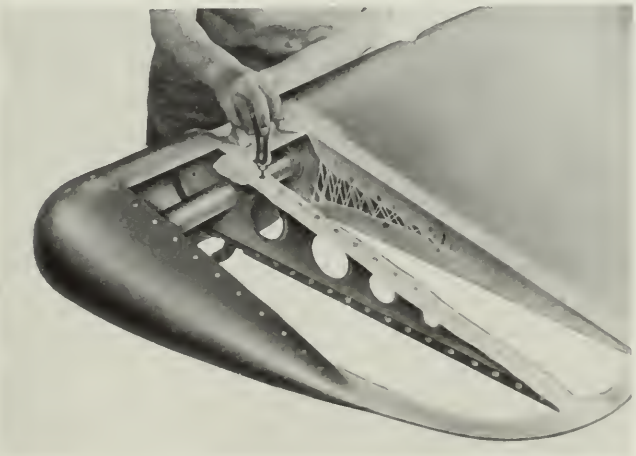


Figure 25—Removing Fabric Retaining Screws

Figure 26—Pulling on Partial Fabric Envelope Cover

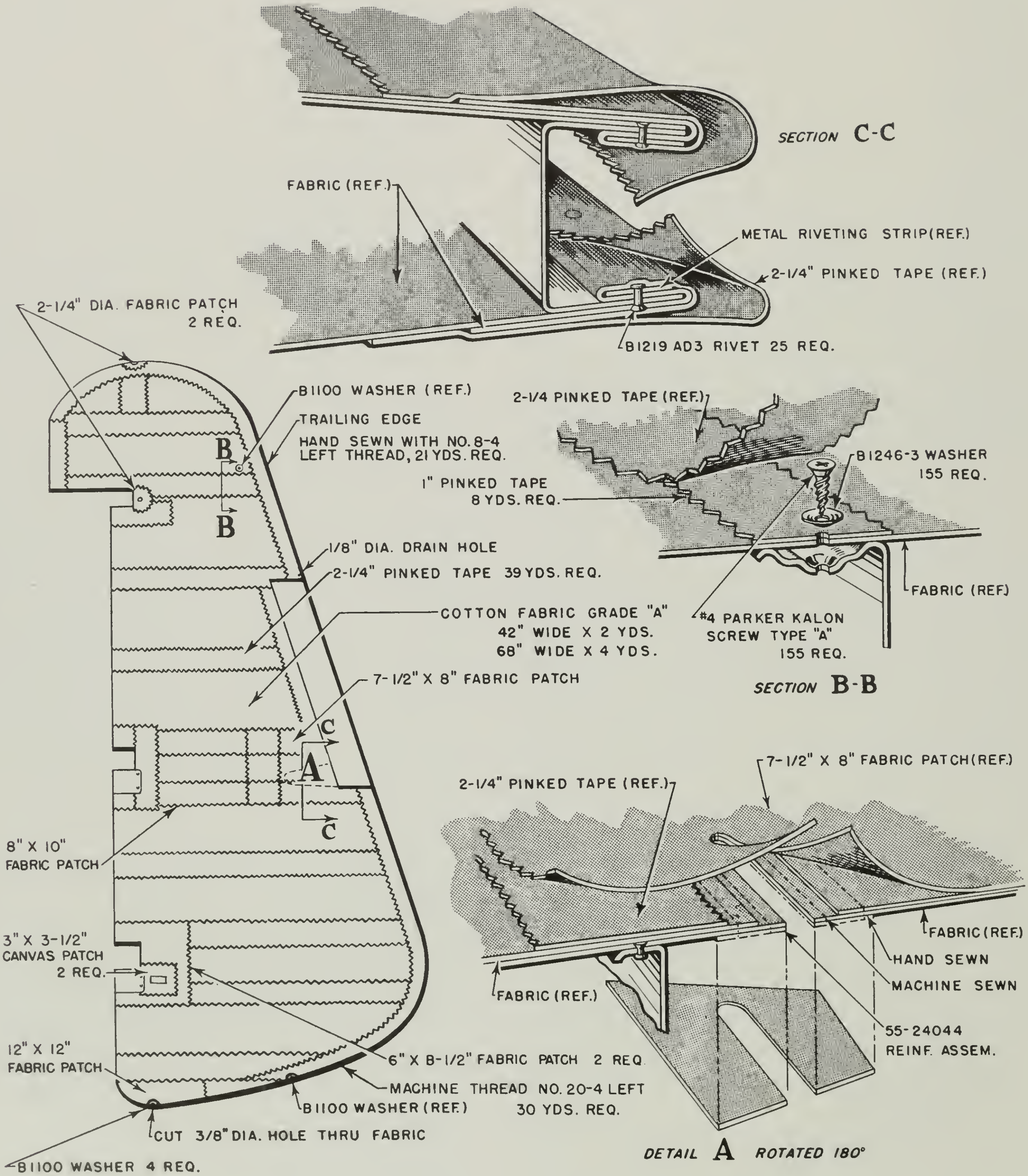


Figure 27—Rudder Fabric Covering Requirements

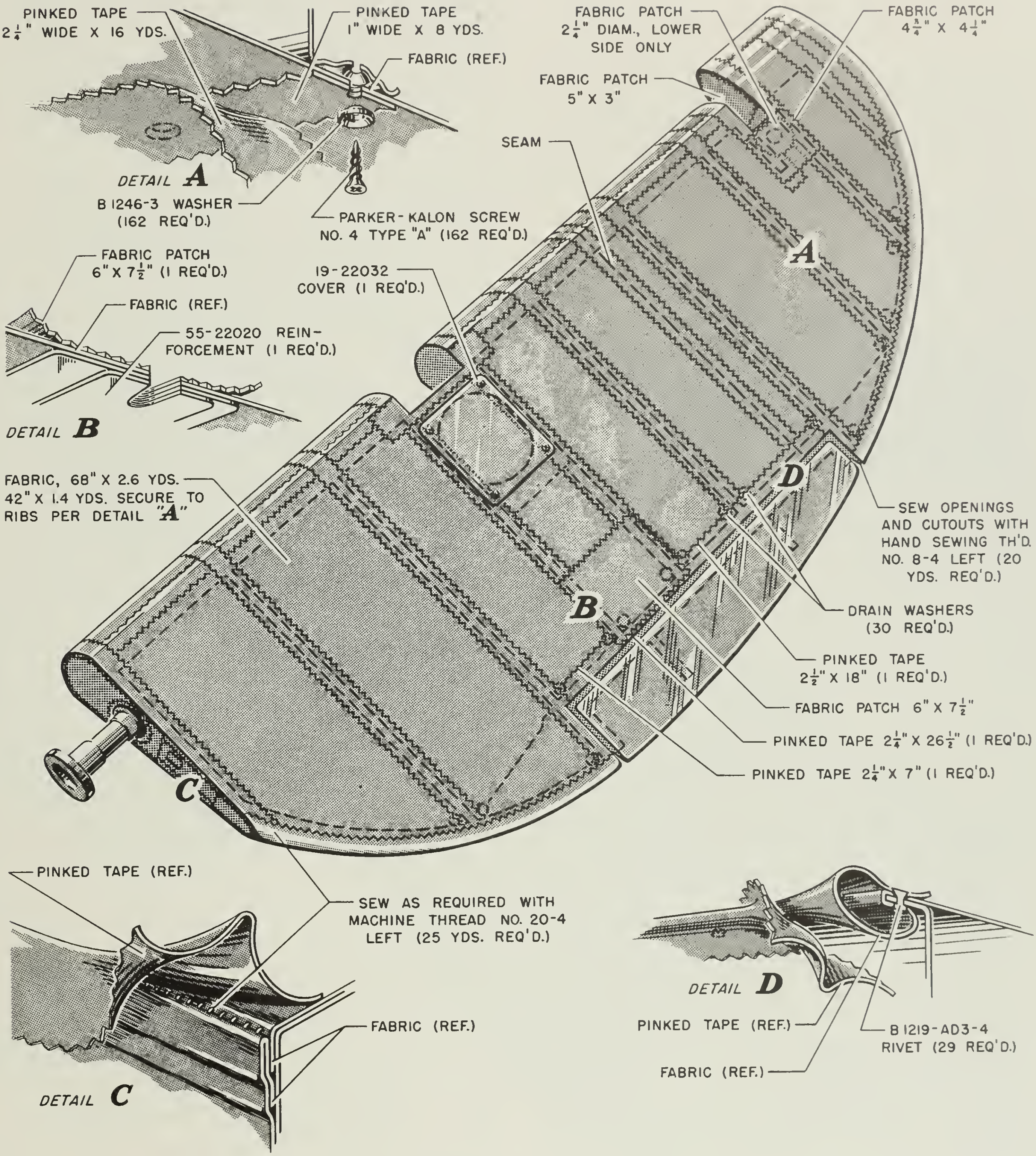


Figure 28—Elevator Fabric Covering Requirements

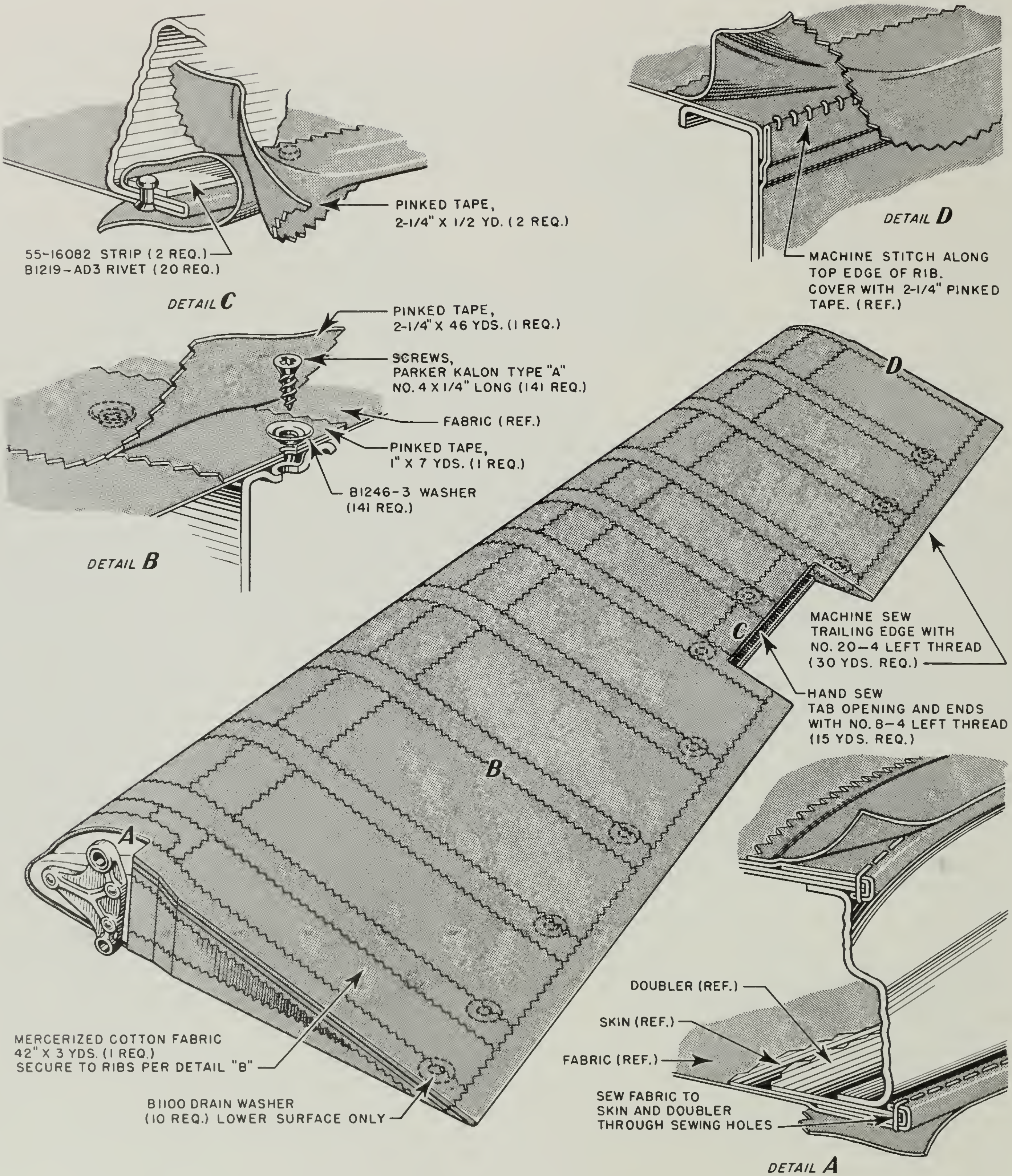


Figure 29—Aileron Fabric Covering Requirements

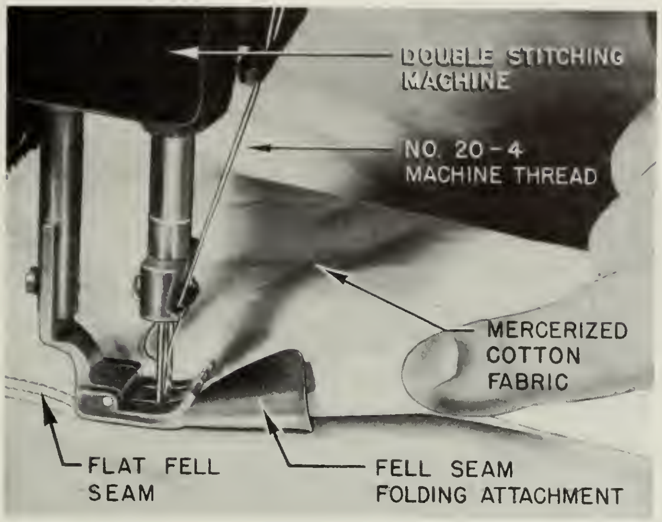


Figure 30—Fabric Double-stitching Machine

30. PREPARING NEW FABRIC ENVELOPE.

The preparation of a new fabric envelope covering for the aileron, rudder, or elevator will differ in some respects, as each structure is of a different shape. These differences will vary the amount of possible machine stitching for each cover. All machine-sewed seams are pointed out on the corresponding illustrations *(see Figures 27, 28, and 29)*. Materials required and their respective quantities are also specified in these diagrams and in the closing paragraphs of this Section. The cover should be prepared and the seams so made that warp threads (parallel to the selvage) lie parallel to the line of flight of the airplane. From the appropriate diagram and the available fabric, determine the number of fabric widths required to cover the span of the surface. Sew them together by means of a double-stitching machine equipped with a double-folding attachment *(see Figure 30)*. If this type of equipment is not available, the folding can be done by hand and the stitching done on a regular single-stitching sewing machine. Care should be taken to ensure that the stitching is uniform and goes through all four layers of the fold. The rows of stitches must be at least 1/16-inch from the edge of the fold and the rows 1/4- to 3/8-inch apart. The cover should be machine-sewed in the form of an envelope. Trailing edge seams that may be machine-sewed are indicated on the re-covering diagrams *(see Figures 27, 28, and 29)*. Pass the fabric about the frame trailing edge, up and over the leading edge, then back on the opposite side to the trailing edge. Pull the fabric evenly to required tautness and pin the edges together at the trailing edge, spacing the pins close. Pull the fabric more in a spanwise direction than in a chordwise direction. With a soft pencil, mark the fabric between and upon the pins. Slip the cover off the small end of the structure and sew two seams along the penciled marks, removing the pins as required. End the machine stitching where indicated on the diagram or at such a place as to permit the covering to be slipped over the narrower end of the frame. Trim the fabric covering to within 1/4-inch of the machine-stitched seam and turn inside out. The envelope cover is now ready to be pulled over the structure.

31. ATTACHMENT OF THE NEW FABRIC ENVELOPE COVER.

Slip the presewed envelope cover over the small end of the structure and pull taut. The application of powdered pumice to the frame will facilitate pulling the fabric along the leading and trailing edges of the structure where the friction is greatest. A small rough rubber sheet may be used as a further aid in pulling or pushing the fabric covering along the leading and trailing edges of the frame *(see Figure 31)*. Press the sheet firmly upon the leading edge and rub the fabric in the required direction. Keep the pull on the leading and trailing edge fabric as nearly equal as possible and work toward the large end of the structure. Pull the fabric taut around the large end of the structure and pin the fabric together with straight pins spaced not more than 2 inches apart *(see Figure 32)*. Pin around the entire unsewed end of the cover and along the trailing edge until the machine stitching is reached. Care must be taken to prevent the

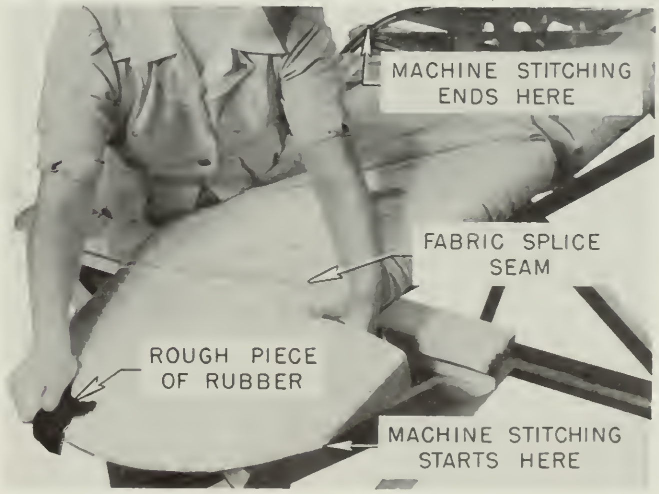


Figure 31—Pulling New Fabric Cover Over Control Surface Structure

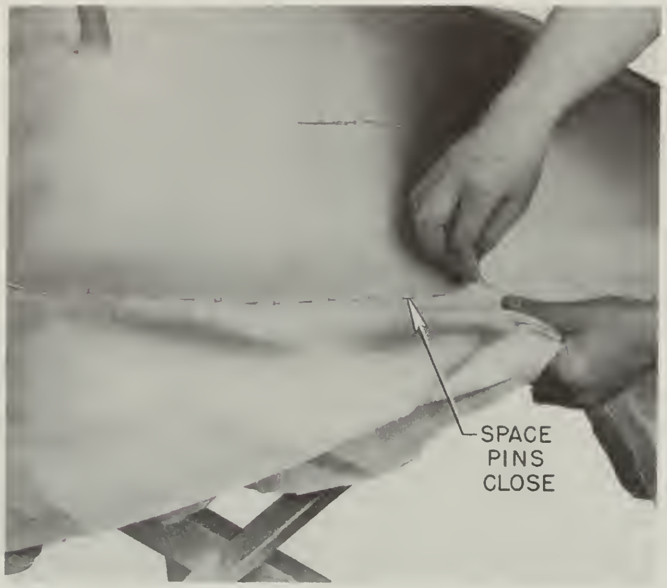


Figure 32—Pinning Unsewed End of Fabric Envelope

fabric from slipping while being pinned and thus causing an uneven pull. The pull or tension on the fabric along the open end and the trailing edge should be uniform. After the unsewed end of the fabric envelope has been pinned in this manner, trim the excess fabric with a pair of regular or pinking shears. Cut the fabric on a line one inch from and parallel to the row of pins *(see Figure 33)*. Apply a coat of dope around all hinge and access cutouts and along the trim tab channel. Allow this dope to dry before removing the pins and stitching the unsewed end. This action retains fabric tautness when the fabric is cut around all such cutouts. Pull the fabric taut around the trim tab opening, and pin the fabric *(see Figure 34)*. Cut the fabric diagonally into each corner of the trim tab opening *(see Figure 35)*. Fold the fabric under the attaching flange of the trim tab channel spar and place the metal riveting strip in alignment with the rivet holes in the flange. Punch holes through the two layers of the fabric at each rivet hole, and insert and drive an AN426-AD3 rivet in each hole *(see Figure 36)*. Trim excess fabric to within 1-1/2 inches of the metal riveting strip; apply one coat of dope to the remaining fabric strip; turn the fabric back around over the riveting strip and up on the fabric surface that covers the trim tab channel spar. This detail is set forth on the re-covering diagram for each surface *(see Figures 27, 28, and 29)*. Obtain or cut all reinforcing and finishing tape required from predoped fabric. Quantities required and dimensions are listed on the re-covering diagrams. If rolls of pinked tape are not available, reinforcing tape, finishing tape, canvas and fabric patches may be cut from doped fabric by the use of a hand pinking machine *(see Figure 37)*. Pinking may also be accomplished with pinking shears if a machine of this type is not available. Apply a coat of dope to the fabric on each rib; and while this dope is still wet, center and pin a length of one-inch predoped reinforcing tape over each rib of the trailing edge. Attach the fabric covering to all ribs by inserting the Parker-Kalon self-tapping screws into the dimpled washers and then into the holes in the recesses provided in each rib capstrip *(see Figure 38)*. Use a pneumatic screwdriver with a No. 1 Phillips head bit if available *(see Figure 40)*. For the proper clutch setting, experiment with identical screws in the same size holes drilled in scrap metal of the same type and thickness.

32. HAND SEWING FABRIC ENVELOPE AFTER ATTACHMENT.

Hand sewing is required along the trailing edge where the machine stitching of the envelope cover terminates, at both ends of the trim tab cutout, and around all hinge and access cutouts. To hand-sew the trailing edge and the previously pinned end of the envelope cover, use a small curved sack needle and a single strand of No. 8, 4-ply cotton hand-sewing thread. Use a baseball stitch, providing a minimum of four stitches per inch. Remove two or more pins as necessary and fold under the excess material remaining after the trimming operation. Start at the point where the machine stitching stops; fold under the material in such a way that, when

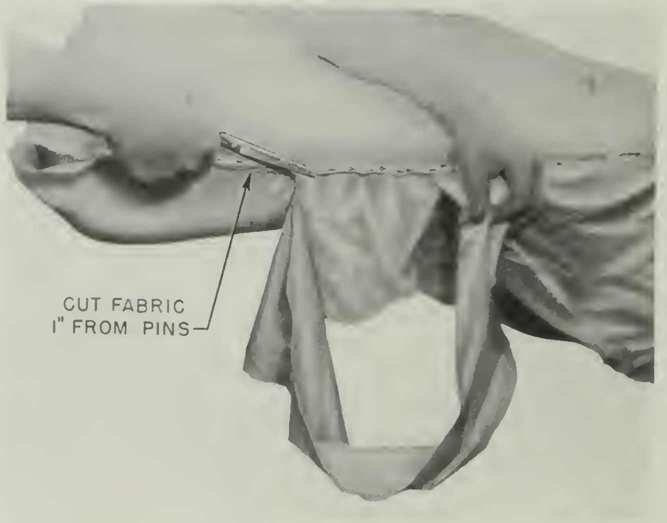


Figure 33—Trimming the Unsewed End of Fabric Envelope After Pinning

the stitching is made, the edges will be pulled tightly together. Knot the sewing every 6 inches. Proceed with the hand sewing until the leading edge is reached *(see Figure 43)*. At both ends of the trim tab cutout, use an overthrow or roll stitch; and starting at the trailing edge, fold the excess fabric under and sew along the edge of the rib until the trim tab channel is reached *(see Figure 39)*. At the hinge bracket cutouts and the access cutouts, small holes in the metal edges are provided for the stitches and a different method of stitching must be utilized. Cut the fabric around the hinge cutout to within 1/2-inch of the metal edge, and fold this fabric under until the rolled edge is even with the metal edge of the cutout. This will form a double thickness around the cutout and a smooth rolled edge for sewing. Sew the fabric edges to the metal edge of the cutout, utilizing the small row of holes provided for the stitching *(see Figure 41)*. Pass the needle through the fabric and the hole in one direction and back through the adjacent hole in the opposite direction for one complete circuit around the edge; then reverse the direction of travel and make another complete circuit so that the sewing appears to be a continuous thread. Similar stitching is utilized at the inboard hinge cutout on the aileron. Here, however, the fabric is folded and this double thickness doubled under the metal edge, forming four thicknesses of fabric to sew through. See the corresponding detail on the aileron re-covering diagram.

33. DOPING FABRIC - GENERAL.

Whenever possible, all doping should be accomplished in accordance with accepted prac-

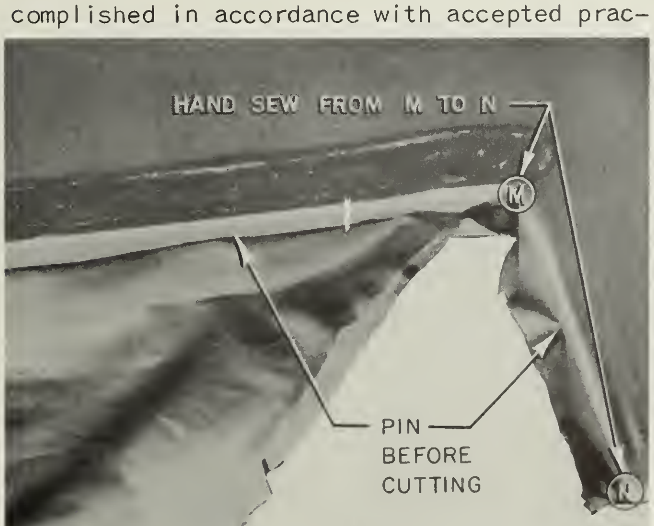


Figure 34—Fabric Pinned at Trim Tab Cutout

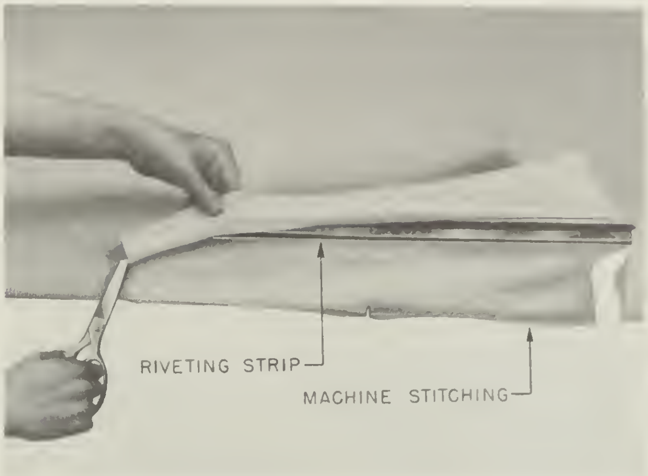


Figure 35—Cutting Fabric at Trim Tab Cutout

tice at room temperatures of from 20° to 23°C (68° to 73°F). Relative humidity should not exceed 50 percent. When small fabric tears are repaired while the control surface is still mounted on the airplane, accomplish all doping operations during the day when the temperature is near 70°F and when there is very little moisture in the air. This is usually between 10 a.m. and 4 p.m. in most localities. If doping is done during the hottest part of the day or if wind is prevalent, the dope will dry too fast and blushing will result. If it is necessary to apply dope under these conditions, add a relative amount of retarder to the dope before application. Retarder slows the drying action of the dope and permits the chemical components of the dope to penetrate the fabric before drying. If blushing is noted after application, remedy this condition by spraying a thin "dry" coat of dope thinner over the area. The same effect may be obtained by wiping the area lightly with a clean cloth saturated with dope thinner. All major doping operations such as fabric section replacement, partial fabric re-covering, and re-covering of the entire surface should be accomplished in a doping room to ensure satisfactory results and a smooth, taut surface. Thin all clear and aluminized dope with lacquer thinner (du Pont formula No. 3585) until the specific gravities, measured in Baumé degrees, are in accordance with the following table.

DOPE	DEGREES BAUMÉ AT 21°C (70°F)
ALUMINIZED	29
CLEAR (BRUSH)	32-1/4
CLEAR (SPRAY)	THIN BRUSH DOPE 1:2

A tolerance of 1/2 Baumé degree is permissible. For the two final spray coats, mix the aluminized dope as follows:

INGREDIENT	MFG. SPEC.	QUANTITY
CLEAR DOPE	231-C-9	1 GALLON
ALUMINUM PASTE	1920	8.33 OZ.
THINNER	3585	TO GRAVITY

34. FABRIC DOPING PROCEDURE.

The doping procedure consists essentially of applying two brush coats of clear nitrate dope, one spray coat of clear nitrate dope, and two spray coats of semipigmented aluminized dope to the fabric-covered control surface. Apply the first brush coat of clear dope to the surface as soon as practical after the new fabric covering has been attached to the structure. This prevents the new material from losing its tautness and flexibility. The newly covered surface should never be allowed to remain overnight without at least one coat of dope applied to it. Apply this first coat of unthinned clear dope with a wide, soft-bristled brush *(see Figure 42)*. All finishing tape along the ribs, the trailing edge, and all seams, and around all cutouts should be applied simultaneously with the second unthinned brush coat of clear dope in accordance with approved methods *(see Figure 44)*. It is particularly important that all surface tape be properly "buried"; that is, all ends should be covered by another tape in such a manner that the slip stream cannot cause the tape to tear free. Equally important is the proper preparation of the ends of the leading and trailing edge tape, particularly those of the latter where they terminate at trim tab cutouts. While the second brush coat of dope is still wet, rub down the pinked edges of all finishing tape, canvas and fabric patches so that a smooth surface will result *(see Figure 45)*. After the second coat of dope has dried thoroughly, lightly sandpaper the entire surface with a fine grade of waterproof sandpaper. Care must be taken not to apply too much pressure between ribs, thus causing stretching of the fabric. Apply the third coat of clear dope with a spray gun *(see Figure 46)*. This spray coat should be wet enough and sufficiently heavy to wet both previous coats and thus flow out the roughness of the undercoats. After this third coat of dope has dried, spray on the two coats of aluminized dope. If a rough surface is attendant after the first spray coat of aluminized dope has dried, sand lightly as outlined above, before spraying on the last coat. The dope finish shall be such as to produce the desired tautness and of sufficient smoothness to be readily cleaned. Doping of all airplane fabric surfaces with semipigmented nitrate dope shall be such as to produce a smooth finish weighing not less that 4 ounces per square yard. Use the above doping procedure whether doping a small patch, a section replacement, or an entire surface. Care must be taken not to apply any more dope than is necessary to produce the required surface, as additional applications of dope add weight to the surface and cause the unbalance limits to be exceeded.

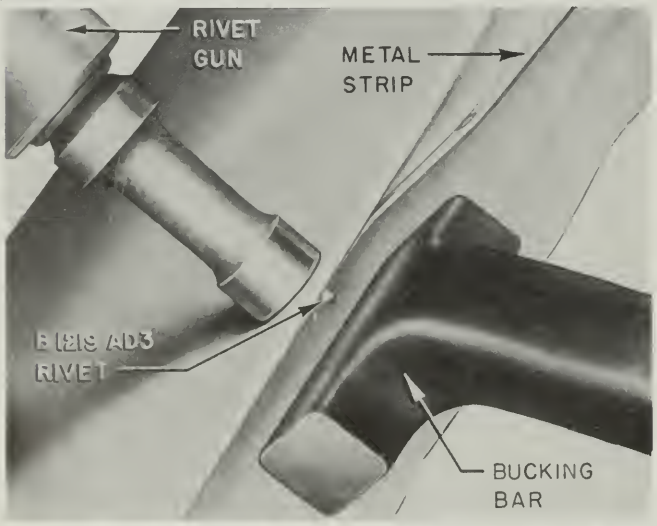


Figure 36—Attaching Fabric Along Trim Tab Channel

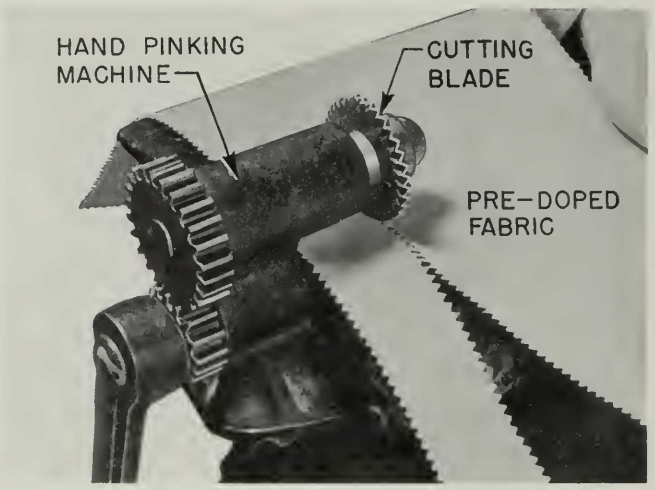


Figure 37—Cutting Tape With Hand Pinking Machine

35. CONTROL SURFACE STATIC BALANCE.

In order to counteract the weight of a control surface, a counterbalance in the form of a heavy lead weight is secured forward of the hinge centerline. In the original construction of the control surfaces for this airplane, an unbalance condition exists; that is, the weight aft of the hinge line exceeds that forward of the hinge line. Because of the relative low speed of this type airplane, this condition is not critical. However, definite unbalance limits are set up which must not be exceeded. *(See Figures 49, 50, and 51.)* The addition of weight to the surfaces in the form of repairs may cause the unbalance limits to be exceeded, and is therefore a possible cause of fluttering or destructive oscillation of the surface in flight. Minor patching of fabric surfaces, which adds negligible weight to the control surface, will not be considered cause for checking the unbalance. It is to be remembered that the nearer the patch is to the hinge line of the surface, the less will be its effect on the unbalance. Large, heavy,

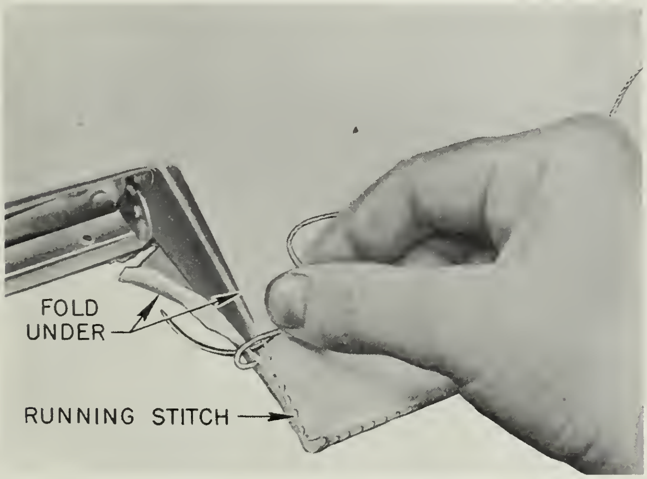


Figure 38—Attaching Fabric to Ribs

or numerous patches, especially those on or near the trailing edge, should therefore be avoided by section replacement or partial re-covering. After any sizeable repairs, most of which will necessitate removing the surface from the airplane in order to perform properly all re-covering and doping operations, check the unbalance before remounting the surface on the airplane. It is necessary that the unbalance be held within the specified limits to avoid possible fluttering of control surfaces at high speeds. Prior to the accomplishment of any balance check, the surfaces must be completely assembled and finished, with balance weights, hinge fittings, trim or booster tabs, and trim or booster tab operating rods all in place. After the surface is thus prepared, use the method outlined in the following paragraph to check the unbalance.

36. DETERMINING CONTROL SURFACE UNBALANCE.

To ascertain whether or not the unbalance of the control surface is within the specified limitations, the surface must be mounted so that it lies in a level, horizontal position and moves freely on its own hinges. Inasmuch as the unbalance need not be checked except after sizeable repairs which will necessitate removal of the surface from the airplane, checking of the unbalance while the surface is mounted on the airplane should not be attempted. A balancing jig, consisting of two welded angle iron standards clamped or bolted to a level table, will adequately serve for this purpose *(see Figure 48)*. The angle iron standards must be so made that the hinge points will be level and will not bind against the structure at any point. The surface when mounted in this manner must be free to move of its own inertia on its own bearings. Place a balance scale that is calibrated in ounces under the trailing edge of the structure at the inboard end of the trim tab cutout on the elevators and ailerons, and at the lower end of the trim tab cutout on the rudder *(see Figures 49, 50, and 51)*. It must be remembered that the weight of any blocks or device placed on the scale to level the structure must be subtracted from the total weight reading on the scale in order to ascertain correctly the unbalance weight of the trailing edge. Refer to the corresponding diagram and to the following table for maximum allowable underbalance conditions *(see Figures 49,50, and 51)*.

STRUCTURE	MAXIMUM SCALE READING
ELEVATOR	2 LBS. 2 OZ.
RUDDER	2 LBS. 4 OZ.
AILERON	13 OZ.

Figure 39—Hand Sewing at Trim Tab Cutout

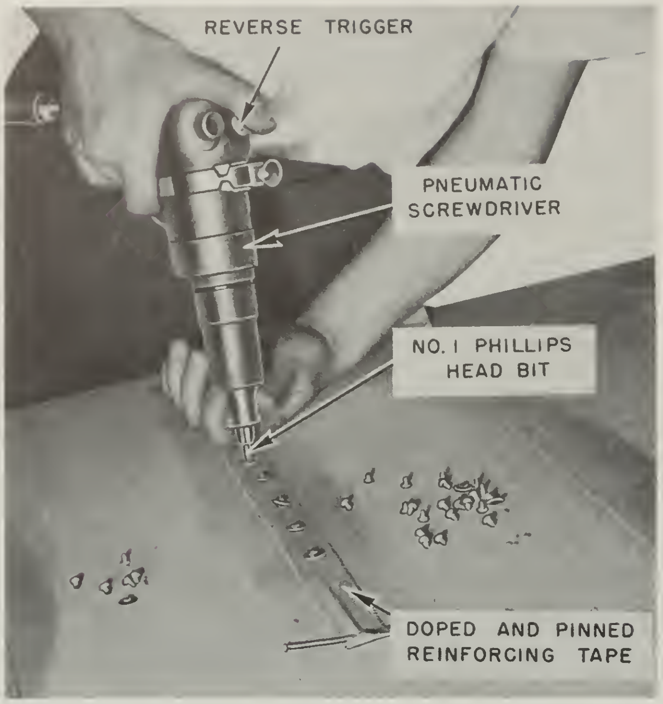


Figure 40—Use of Pneumatic Screwdriver to Attach Fabric to Ribs

If the weight exceeds these limits, the balance is not satisfactory and the surface must not be used until corrective action has been taken. Methods of correcting static unbalance are set forth in the following paragraphs.

37. CORRECTING UNBALANCE WHEN ABOVE LIMIT.

If, after following the instructions in the preceding paragraph, the unbalance weight exceeds the maximum allowable limit, this condition may often be corrected by the addition of weight forward of the control surface hinge

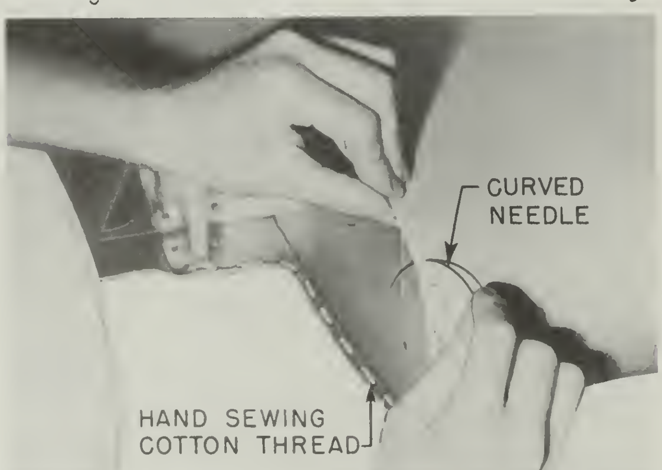


Figure 41—Hand Sewing Fabric at Hinge Cutout

Figure 42—Applying Brush Coat of Dope to Fabric

centerline. Addition of weight in the form of long, heavily doped patches applied to the leading edge of the aileron or to the balance weight housing of the rudder or elevator will sometimes bring the unbalance weight within the limit. To apply such a balance patch to the aileron, apply a heavy coat of unthinned dope to the leading edge of the aileron. Place the fabric patch in position over the wet dope and rub down firmly until the dope has penetrated the fabric *(see Figure 47)*. Allow to dry before applying any additional coats of dope. If necessary, a total of six coats of dope may be

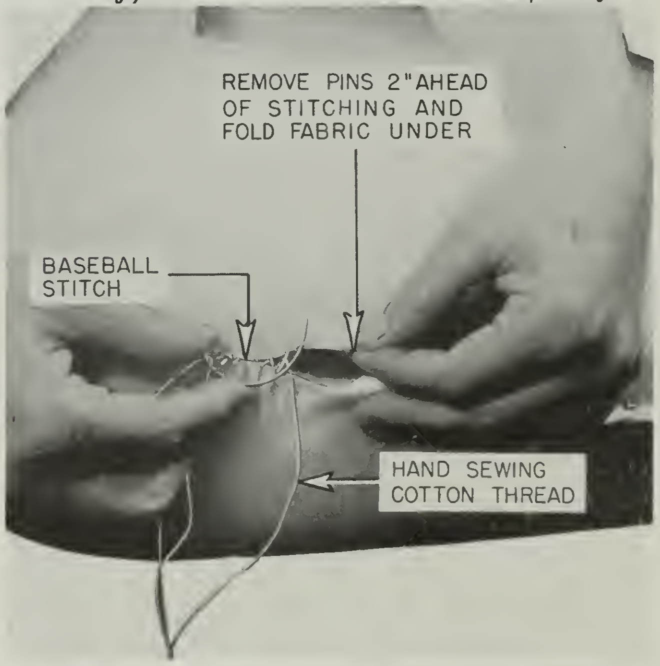


Figure 43—Hand Stitching the Unsewed End of Fabric Envelope

Figure 44—Applying Finishing Tape to Fabric

applied to each patch, and a total of three long patches may be applied to the same area. Continue adding patches and dope to the leading edge until the reading on the balance scale is within the specified limitations. If, after the application of three patches to the leading edge, the unbalance is still beyond the limit because of numerous patches aft of the spar, replace the entire affected fabric section or sections if feasible; otherwise, re-cover the entire surface and recheck the static balance. The procedure for the rudder and elevator follows that outlined above with the exception that the balance patches are applied to the balance weight housing instead of the leading edge *(see Figure 52)*.

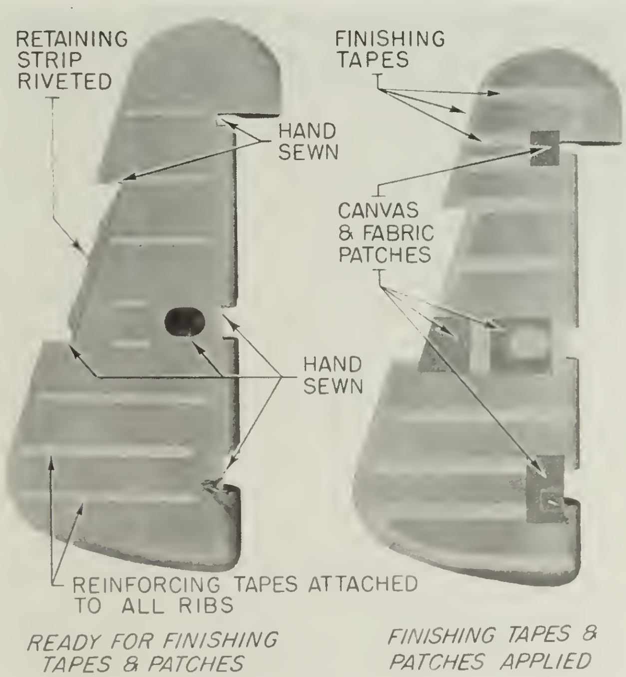


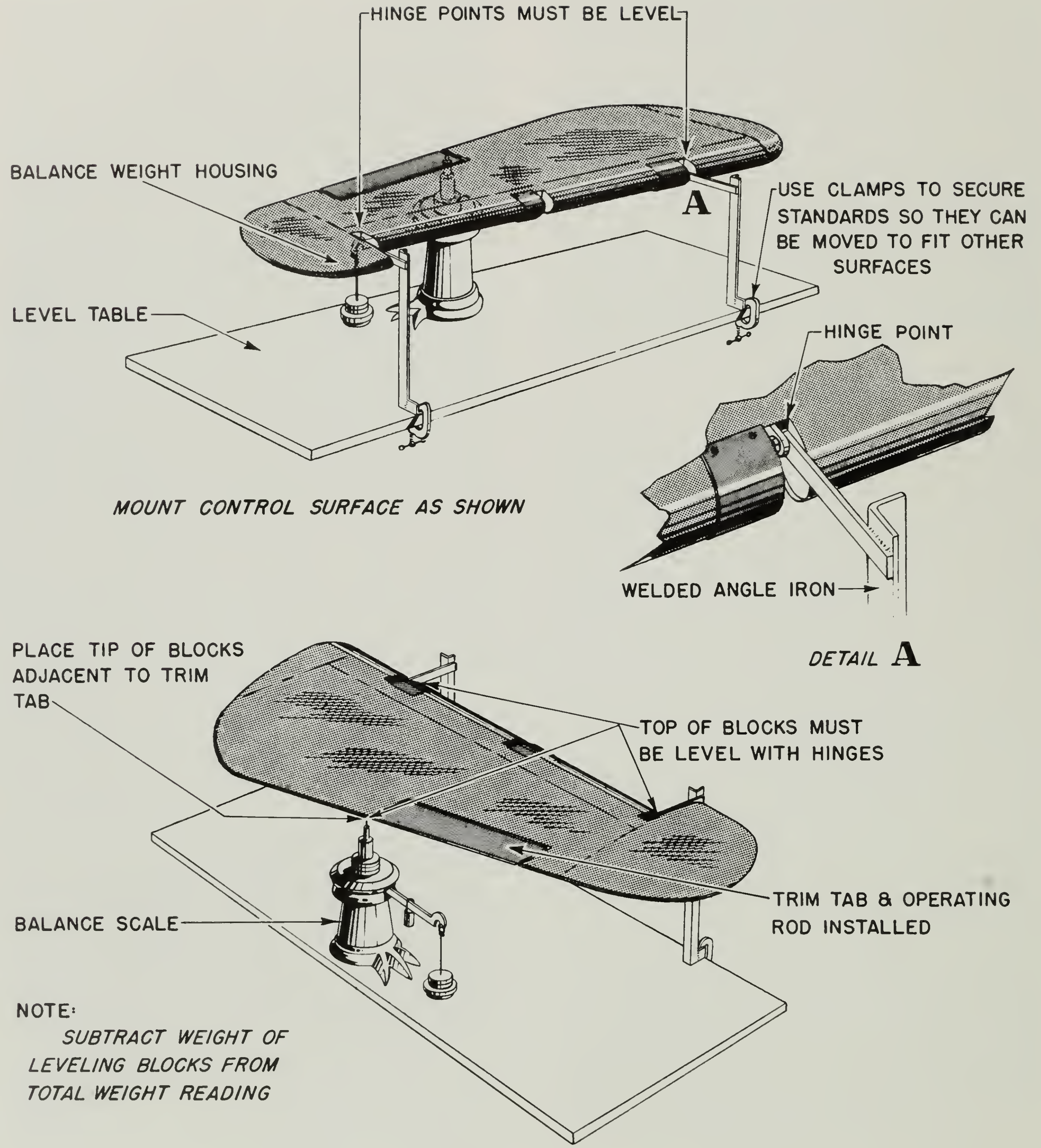
Figure 45—Fabric Envelope Prior to First and Second Coats of Dope

Figure 46—Applying Spray Coats of Aluminized Dope to Fabric

38. METAL AND FABRIC REPAIR TOOLS.

In addition to the standard sheet metal cutting, forming, bending, and riveting tools, all or part of the following tools will be required for structural repairs to all members and for repair to the fabric covering of the elevator, rudder, and ailerons.

Figure 47—Applying Leading Edge Balance Patch to Aileron



CHECK BALANCE AT TRAILING EDGE

Figure 48—Determining Control Surface Unbalance

TOOL	REMARKS
EQUIPMENT	DOPE SPRAYING
SHEARS	PLAIN AND PINKING
MACHINE, SEWING	*(See Figure 30.)*
MACHINE, PINKING	*(See Figure 37.)*
KNIFE	
BRUSH, DOPE	WIDTH TO SUIT
NEEDLES	CURVED AND STRAIGHT TO SUIT
SCREWDRIVERS	NO. 1 AND NO. 2 PHILLIPS

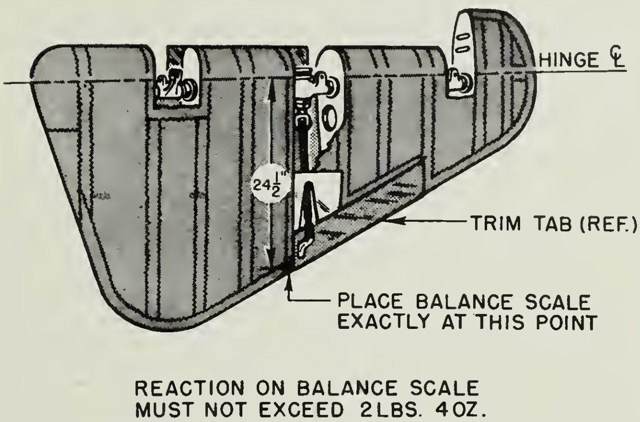


Figure 49—Rudder Unbalance Limits

39. METAL STRUCTURE REPAIR MATERIALS.

The following is a list of materials needed to repair properly the metal structures of elevator, rudder, ailerons, and landing flaps.

MATERIALS	REMARKS
SHEET, 24ST ALCLAD	.020 TO .040 INCH THICK
RIVETS, AN442-AD4	FLAT HEAD
RIVETS, AN426-AD4	COUNTERSUNK
RIVETS, AN426-AD3	COUNTERSUNK
RIVETS, B1227-AD4	BRAZIER HEAD
RIVETS, LS1127-4-4	CHERRY BLIND RIVETS
RIBS, REPLACEMENT	*(See Figures 2, 3, 5, and 7.)*

40. RUDDER COVERING MATERIALS.

The following materials are required for the complete re-covering of the rudder. *(See Figure 27.)* Accomplish all patching or partial re-covering by the use of the same materials in smaller quantities. All patch materials should conform to the same specifications as required for the original covering of the structure.

MATERIAL	AMOUNT REQUIRED	AAF SPEC.
FABRIC, MERCERIZED COTTON	42 IN. X 2 YDS. 68 IN. X 4 YDS.	AN-CCC-C-399
TAPE, PINKED	2-1/4 IN. X 40 YDS.	6-62B
TAPE, PINKED	1 IN. X 8 YDS.	6-62B
CANVAS, 10 OZ.	3 IN. X 3-1/2 IN. (2 REQ.)	CCC-D-771 TYPE II
PATCH, FABRIC	2 IN. X 9-1/2 IN.	AN-CCC-C-399
PATCH, FABRIC	4 IN. X 18 IN. (2 REQ.)	AN-CCC-C-399
PATCH, FABRIC	7-1/2 IN. X 8 IN.	AN-CCC-C-399
PATCH, FABRIC	8 IN. X 10 IN.	AN-CCC-C-399
PATCH, FABRIC	6 IN. X 8-1/2 IN. (2 REQ.)	AN-CCC-C-399
PATCH, FABRIC	12 IN. X 12 IN.	AN-CCC-C-399
PATCH,	2-1/4 IN. DIAM. (2 REQ.)	AN-CCC-C-399
REINF. ASSEM. 55-24044	1 REQ.	
WASHER, B1100	4 REQ.	
WASHER, B1246-3	155 REQ.	
SCREW, PARKER-KALON, NO. 4, PHILLIPS HEAD, TYPE A, C'SUNK	155 REQ.	
SCREW, B1286-832-7	12 REQ.	
SCREW, B1287-832-6	2 REQ.	
THREAD, MACHINE, NO.20, 4-PLY, LEFT	30 YDS.	V-T-276b, TYPE IBI
THREAD, HAND-SEWING, NO.8, 4-PLY, LEFT	21 YDS.	V-T-276b, TYPE IIIB
DOPE, CELLULOSE NITRATE, CLEAR	AS REQUIRED *(See corresponding par.)*	AN-TT-D-514
PASTE, ALUMINUM-PIGMENT, AIRCRAFT	AS REQUIRED *(See corresponding par.)*	AN-TT-A-461

41. ELEVATOR COVERING MATERIALS.

Materials required for the complete re-covering of each elevator section are listed below. For location and dimensions of tapes and patches, refer to the elevator re-covering diagram. *(See Figure 28.)*

MATERIAL	AMOUNT REQUIRED	AAF SPEC.
FABRIC, MERCER-IZED COTTON	42 IN. X 1.4 YDS. 68 IN. X 2.6 YDS.	AN-CCC-C-399
TAPE, PINKED	2-1/4 IN. X 17-1/2 YDS.	6-62B
TAPE, PINKED	1 IN. X 8 YDS.	6-62B
PATCH, FABRIC	4-5/8 IN. X 4-5/8 IN.	AN-CCC-C-399
PATCH, FABRIC	4-1/4 IN. X 4-3/4 IN. (2 REQ.)	AN-CCC-C-399
PATCH, FABRIC	1-3/4 IN. X 9-1/8 IN.	AN-CCC-C-399
PATCH, FABRIC	5-1/8 IN. X 9-1/8 IN.	AN-CCC-C-399
PATCH, FABRIC	8 IN. X 9-1/2 IN.	AN-CCC-C-399
PATCH, FABRIC	4-3/4 IN. X 2-5/8 IN.	AN-CCC-C-399
PATCH, FABRIC	2-1/2 IN. X 2-3/8 IN.	AN-CCC-C-399
PATCH, FABRIC	7-1/2 IN. X 6-1/4 IN.	AN-CCC-C-399

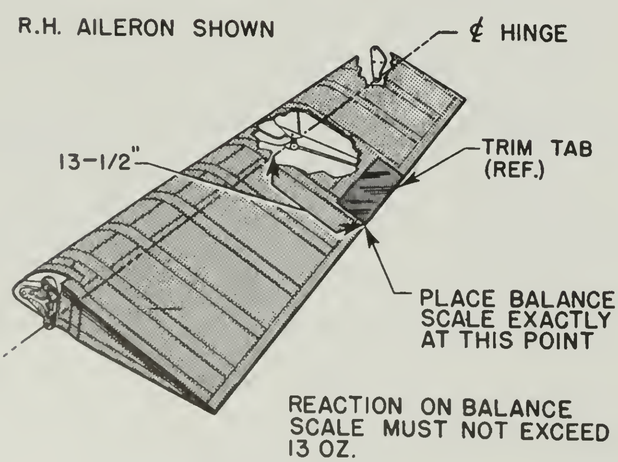


Figure 50—Aileron Unbalance Limits

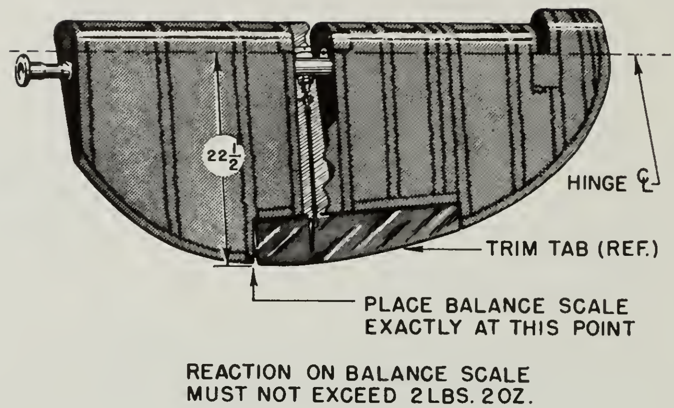


Figure 51—Elevator Unbalance Limits

MATERIAL	AMOUNT REQUIRED	AAF SPEC.
PATCH, FABRIC	6 IN. X 7-1/2 IN.	AN-CCC-C-399
PATCH, FABRIC	2-1/4 IN. DIA.	AN-CCC-C-399
REINF. ASSEM. 55-22020	1 REQ.	
WASHER, B1100	30 REQ.	
WASHER, B1246-3	162 REQ.	
SCREW, PARKER-KALON, NO. 4, PHILLIPS HEAD, TYPE A, C'SUNK	162 REQ.	
SCREW, B1286-832-7	8 REQ.	
SCREW, B1287-832-6	2 REQ.	
THREAD, MACHINE, NO. 20, 4-PLY, LEFT	25 YDS.	V-T-276b, TYPE IBI
THREAD, HAND-SEWING, NO. 8 4-PLY, LEFT	20 YDS.	V-T-276b, TYPE IIIB
DOPE, CELLULOSE NITRATE, CLEAR	AS REQUIRED *(See corres. Par.)*	AN-TT-D-514
PASTE, ALUMI-NUM-PIGMENT, AIRCRAFT	AS REQUIRED *(See corres. Par.)*	AN-TT-A-461

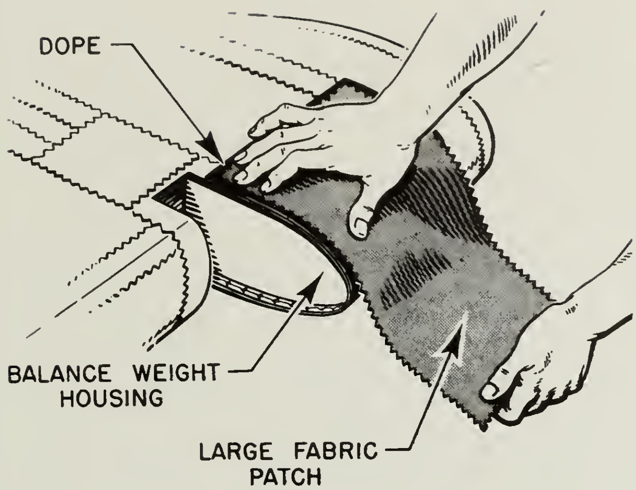


Figure 52—Applying Balance Patch to Elevator and Rudder

42. AILERON COVERING REQUIREMENTS.

The materials listed in the following table are required for the complete re-covering of each aileron. For location and dimensions of tape and patches, refer to aileron re-covering diagram. *(See Figure 29.)* Accomplish all patching or partial re-covering by the use of the same materials in the required smaller quantities. All patch materials should conform to the same specifications as required for the original covering of the structure.

MATERIAL	AMOUNT REQUIRED	AAF SPEC.
FABRIC, MERCERIZED COTTON	42 IN. X 3 YDS.	AN-CCC-C-399
TAPE, PINKED	2-1/4 IN. X 46-1/2 YDS.	6-62B
TAPE, PINKED	1 IN. X 7 YDS.	6-62B
WASHER, B1100	10 REQ.	
WASHER, B1246-3	141 REQ.	
SCREW, PARKER-KALON, NO. 4, PHILLIPS HEAD TYPE A, C'SUNK	141 REQ.	
THREAD, MACHINE, NO. 20, 4-PLY, LEFT	30 YDS.	V-T-276b, TYPE IBI
THREAD, HAND-SEWING, NO. 20, 4-PLY, LEFT	15 YDS.	V-T-276b, TYPE IIIB
DOPE, CELLULOSE NITRATE, CLEAR	AS REQUIRED *(See corresponding Par.)*	AN-TT-D-514
PASTE, ALUMINUM-PIGMENT, AIRCRAFT	AS REQUIRED *(See corresponding Par.)*	AN-TT-A-461

SECTION 5

FUEL AND OIL SYSTEM

1. GENERAL.

The two removable fuel tanks are housed within the wing centersection structure and are held securely in position by three felt-padded metal straps around each tank. *(See Figure 1.)* The tanks are installed between the front and rear spars in the wing centersection structure, and access to them is gained by removing the fuel tank doors which are bolted to the lower side of the centersection structure. The oil tank, located at the top of the engine accessory compartment, is supported on the upper tubes of the engine mount structure near the firewall. *(See Figure 5.)* Two felt-padded metal straps support the tank between two upper cross tubes attached to the engine mount structure.

2. STRUCTURAL DESCRIPTION.

The fuel and oil tanks are each constructed of two power-formed aluminum alloy sections, or shells, welded together. The fuel tanks have also a number of baffles of 52SO material, spot-welded across their inner widths. A vented boxlike compartment of 52S-1/2H material is formed over the fuel sump between two centrally located baffles to retain a volume of fuel over the sump. *(See Figure 2.)* The oil tank has no internal baffles but depends upon the shape of its outer shell for its strength. *(See Figure 4.)* A tubular, vertical compartment through the center of the tank serves as an oil warming compartment. *(See Figure 3.)* This seamless tube is fabricated of 52SO aluminum alloy and is welded into position. All fittings on the fuel and oil tanks are cast from 195-T6 aluminum alloy.

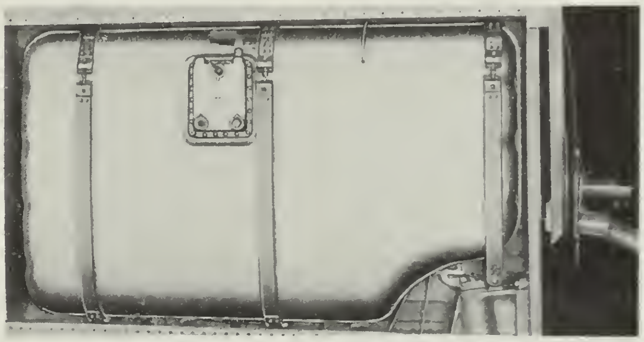


Figure 1—Fuel Tank Supports

3. GENERAL REPAIR.

Repairs to the fuel and oil tanks consist

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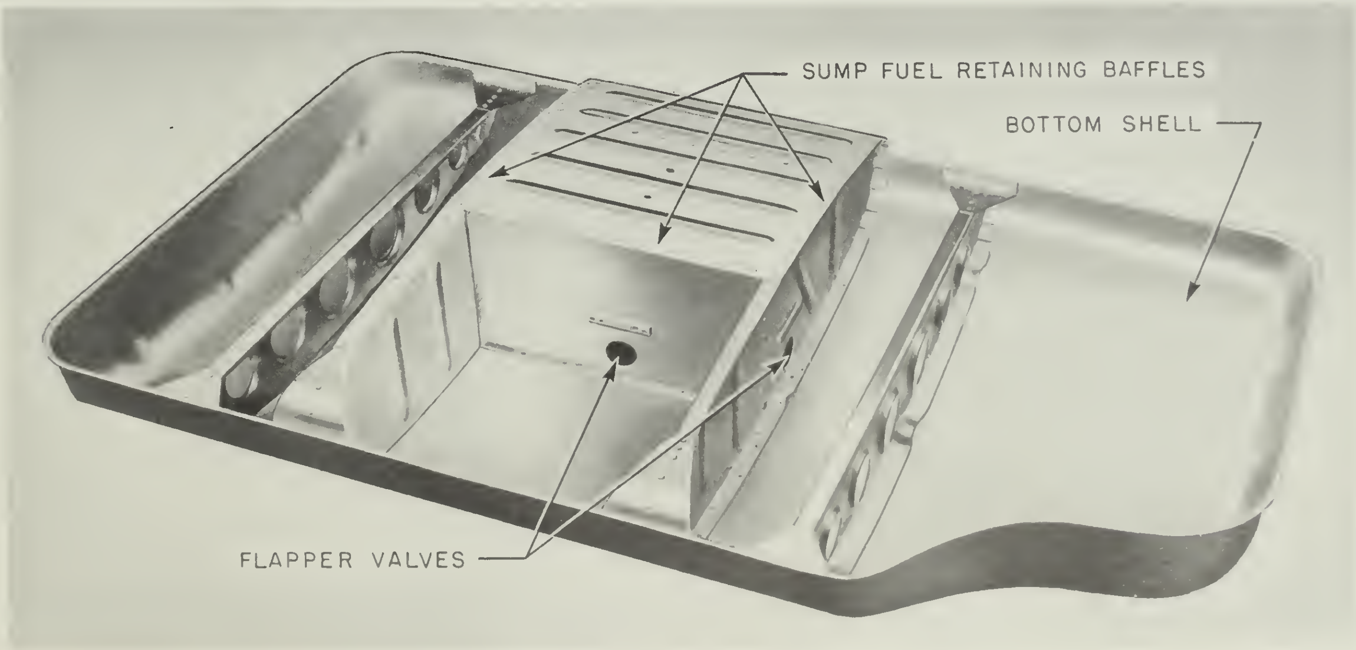


Figure 2—Fuel Tank Construction

of smoothing out negligible damage, sealing small cracks, welding split seams, and welding inserts into holes or large cracks. Prior to welding or applying an open flame of any character, steam the inside of the tank thoroughly for at least three hours, or carefully flush with fire extinguisher fluid (carbon tetrachloride). A one-hour period is sufficient for the oil tank. Prior to welding fuel tank areas located between the outermost baffles, remove the synthetic rubber flapper valves located within the central chamber. These valves are subject to distortion when in the proximity of heat. Access to the valves may be gained by removing the fuel sump.

4. LOCATING SMALL CRACKS.

The presence of minor seepage and vapor leaks in the fuel and oil tanks may be ascertained by air pressure testing. Support the tank in a manner equivalent to the original installation before starting the test. Accessibility

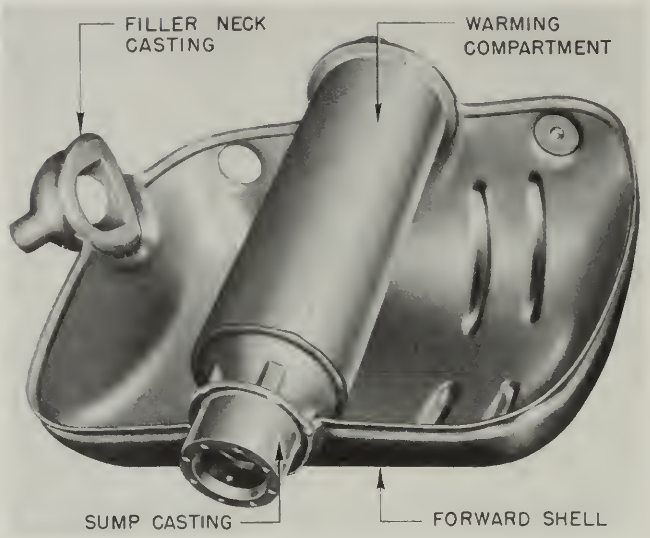


Figure 3—Oil Tank Construction

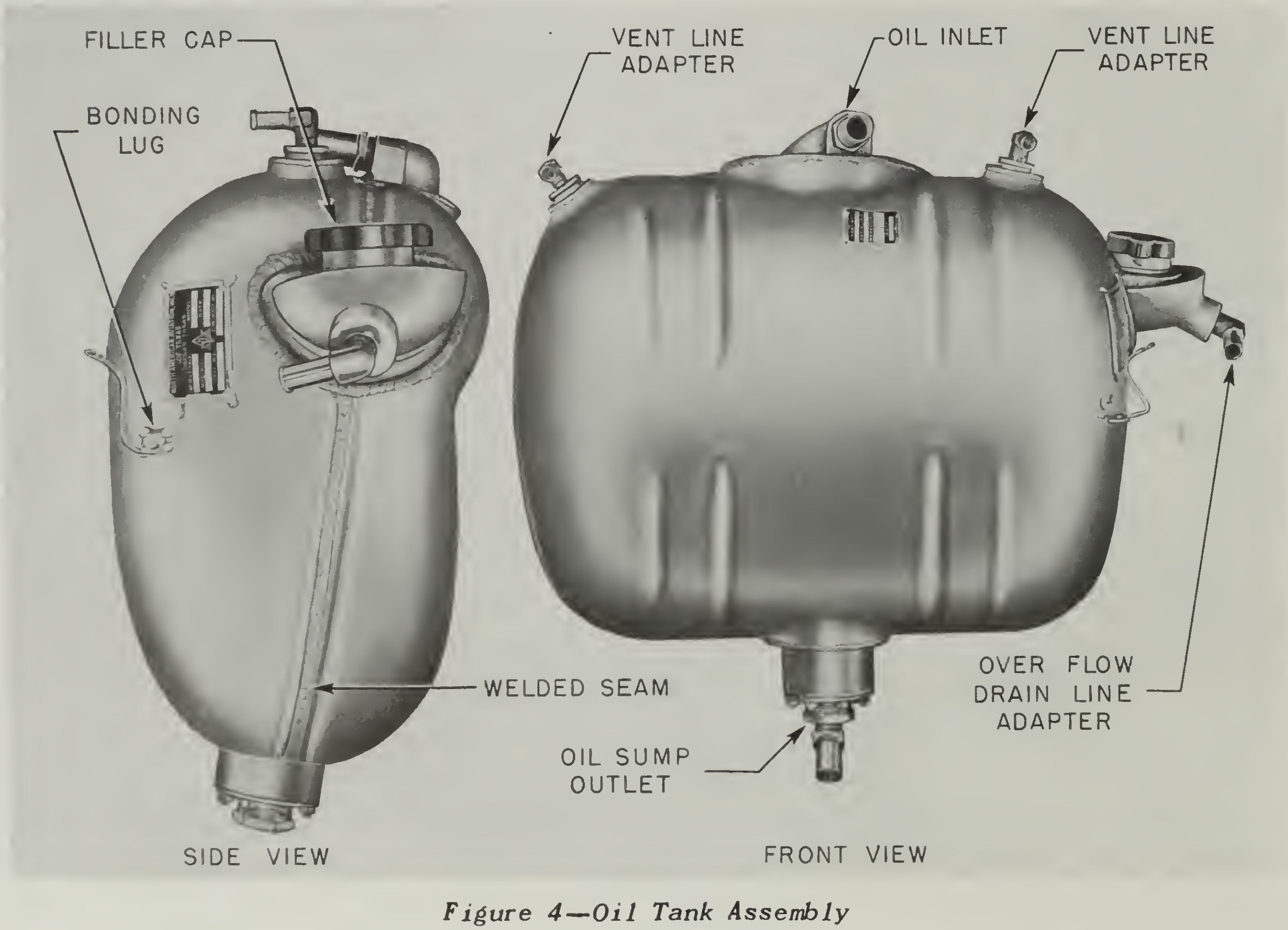
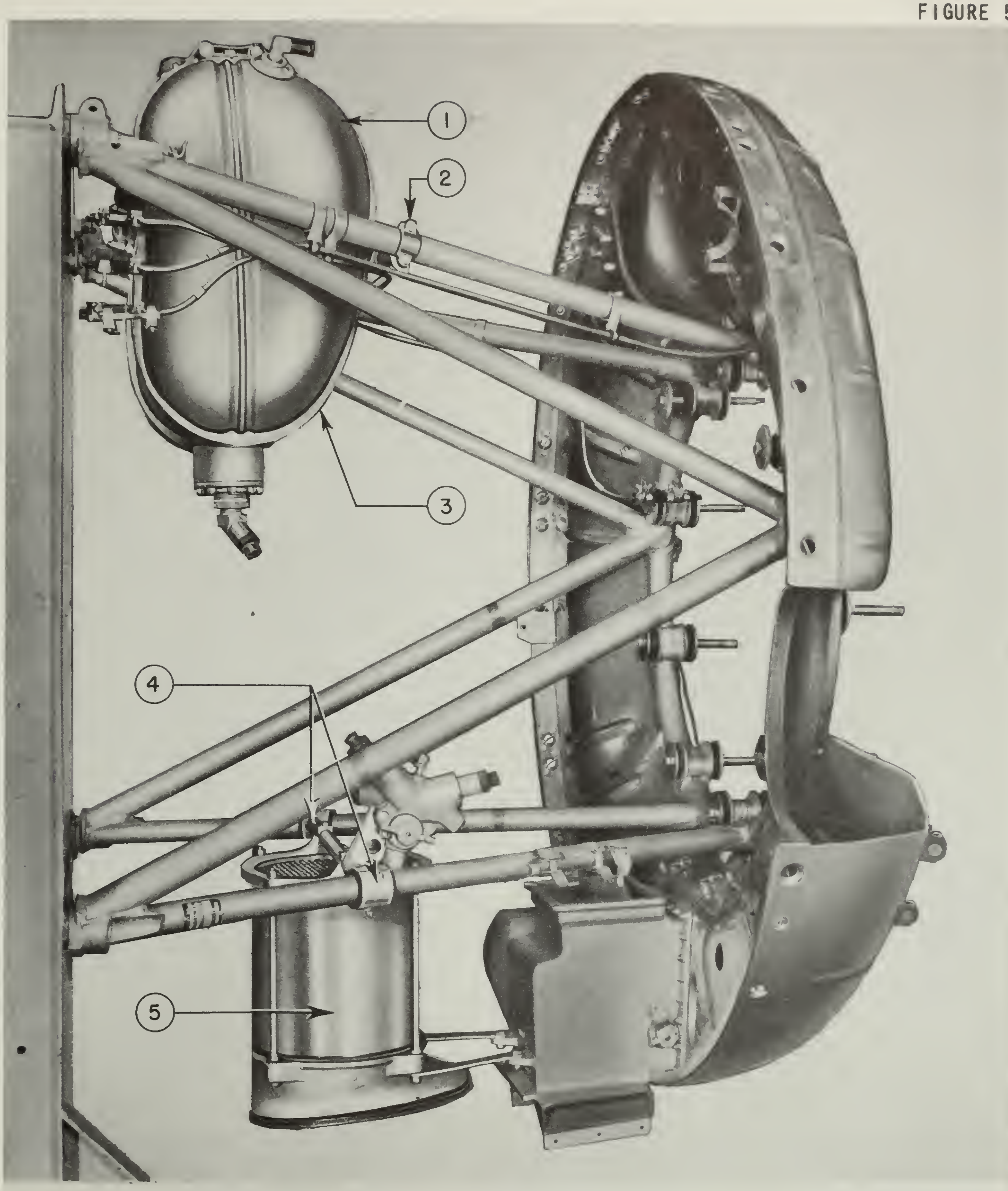


Figure 4—Oil Tank Assembly



1. OIL TANK
2. FRONT SUPPORT
3. PADDED METAL STRAP
4. OIL COOLER SUPPORTS
5. OIL COOLER

Figure 5—Oil Tank and Oil Cooler Installation

of the oil tank seams and fittings while installed permits the use of the original installation supporting straps for this purpose. Supporting the fuel tanks may be accomplished by sawing out pieces of lumber (2 inches thick, 8 inches wide, 42 inches long) to the contour of the tank, padding the edges which will be in contact with the tank with felt, and securing the ends together with strap hinges and hasps as illustrated. *(See Figure 6.)* Apply and maintain an air pressure of not more than 5 lbs./sq.in. to the oil tank or 3-1/2 lbs./sq.in. to each fuel tank. Connect the air supply hose to the tank sump or filler opening after plugging tightly any remaining outlets. Mix one cup of water with 1/2-cup of liquid castile soap; and using a soft-bristled brush, apply this solution to all seams or suspected areas. The formation of a soap bubble indicates a leak. If a soap bubble is found, mark the location and proceed with the examination until the entire tank has been covered. After all leaks are located and marked, release the air pressure and proceed with repairs as outlined in the following paragraphs.

5. SEALING SMALL CRACKS.

Minor seepage and vapor leaks, caused by minute cracks, may be repaired by sealing with a suitable synthetic rubber mixture. Before applying sealant, clean all dirt and grease from the area to be treated and brighten with steel wool. The sealant used must be unaffected by fuel and possess good adhesive qualities.

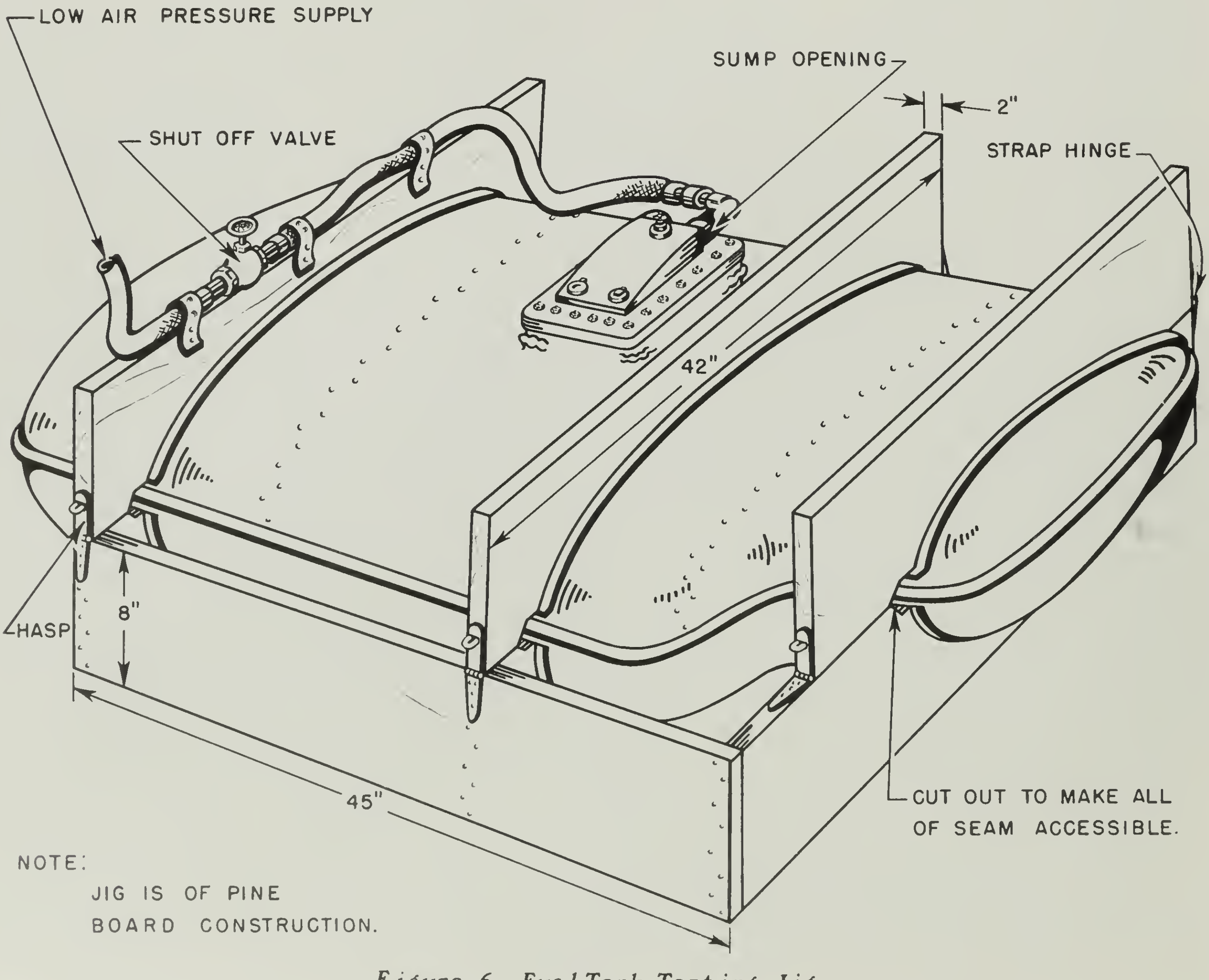


Figure 6—Fuel Tank Testing Jig

Dolphin No. 1625 water and gasoline resisting cement, made by the Dolphin Paint and Varnish Co., Toledo, Ohio, is recommended. Dolphin cement is supplied in a paste form and is applied by pressing it into the cracks with the fingers or thumb. *(See Figure 7.)* Fillet out well to a feather edge and remove all excess cement. The tank may be used immediately after repair. This type of repair will last a considerable length of time; but if the crack has a tendency to enlarge, it is advisable to replace the synthetic compound by means of a welded insertion-type patch as set forth in the following paragraphs. If Dolphin cement is not immediately available, du Pont Cavalprene Cement, manufactured by E. I. du Pont de Nemours & Co., Fairfield, Connecticut, may be used; but sufficient time must be allowed for drying before the tank may be used. Test for sufficiency of repair as outlined in a subsequent paragraph. If insufficiency of repair is indicated, apply sealant to the inner surface of the tank in a similar manner, but do not use steel wool on the tank interior. Access to the tank interior is gained by removing the tank sump. If access cannot be gained by this method, the leak should be repaired in accordance with the instructions in the following paragraphs.

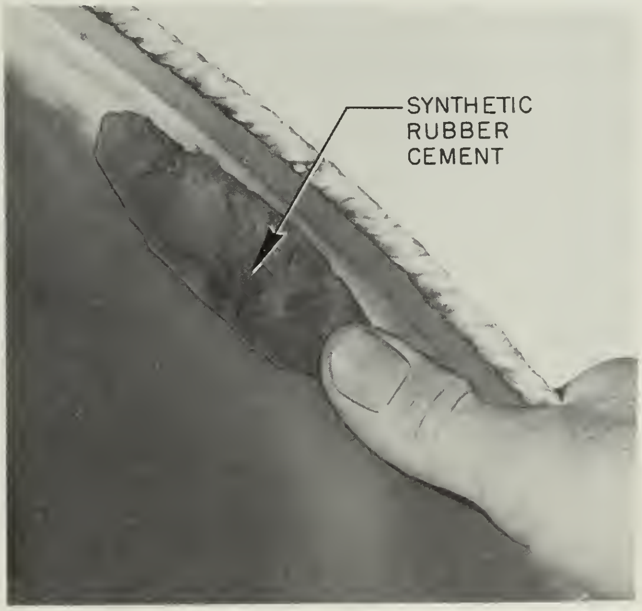


Figure 7—Sealing Small Cracks

6. REMOVING DENTS.

All dented areas in the fuel and oil tanks should be restored to the original contour of the tank shell. This may be accomplished by using a leather mallet and striking the dented area from the inside. This method may be used only when it is possible to gain access to the interior of the tank. If this method is not practical, dents may sometimes be pulled back into shape by welding a 2SO welding wire to the lowest point of the dent and by exerting a steady pull on this wire while applying a torch flame evenly over the dented area. If this method is successful, no further repair is necessary. Snip off the welding wire and file the sharp point. However, if this "pulling-wire" method fails, or if residual cracks or abrasions are attendant after re-forming, cut out the damaged area and treat as set forth in the corresponding subsequent paragraphs.

7. SPLIT SEAM REPAIR.

Dripping or running leaks caused by cracked welds along the seams may be remedied by fusing the crack together by means of a welding torch. *(See Figure 8.)* Hydrogen welding facilities are desirable, but oxyacetylene equipment may be used with the application of special technique. Use a tip a size larger than would ordinarily be used with steel of the same gage and adjust the torch to produce a slightly carbonizing flame. The torch should produce a low velocity bluish flame and not be "blowy." Use 2SO welding wire in conjunction with suitable fluxes. Concluding welding, neutralize all welding fluxes by flushing with a warm solution of 5 percent sulphuric acid and water.

Figure 8—Welding Split Seams

8. LARGE CRACK AND HOLE REPAIR.

Holes and large cracks which may be rounded into holes may be repaired by welding a square, a circular, or an ovaloid insert into a flanged opening. Such repairs are illustrated in this section. *(See Figures 10 and 11.)* Proceed as follows: If the hole is somewhat regular to begin with, file it out to form a clean circle or square, whichever is the most suitable. To prevent filings from entering the tank interior, turn the tank upside down, reverse the ends of the file, and file downward. Next, turn the tank to its original position and flange the edge of the hole slowly upward, as illustrated, by means of a metal bar or pipe and a heavy hammer. *(See Figure 10.)* Form an insert of 2SO, 3SO, or 52SO material over a wooden block. The material should be cut to such size and shape as to permit the insert, upon completion of forming, to be pressed easily into the flanged opening in the tank. Press the insert into position and apply a weld around the top. Completed welded insert repairs generally assume one of the three common shapes illustrated. *(See Figure 11.)* Concluding welding, neutralize all welding fluxes by flushing with a warm solution of 5 percent sulphuric acid and water. Large holes, long tears, or lengthy cracks may be repaired in a similar manner, the insert being shaped to suit the condition. The repair of holes by welding a flat patch over the damage is not recommended, as tests have proven that such repairs fail under prolonged service. However, a flat welded patch may be used in the repair of holes in the oil tank hopper, as this tube is merely a partition between the warm and cold oil. The hopper is easily removed by filing the weld around the hopper inlet plate and pulling the assembly outward.

9. HYDRAULIC OIL TANK REPAIRS.

Inasmuch as the construction of the hydraulic oil tank is similar to that of the lubricating oil tank, all repairs set forth for the latter will be applicable. *(See Figure 9.)* The ends of the tank may be removed by filing the welds around the periphery of the tank. Drain and flush the tank with carbon tetrachloride before attempting any repairs.

10. TESTING.

Upon completion of repairs, thoroughly clean and flush the inside of the tank, using a warm solution of 5 percent sulphuric acid and water. Steam the interior of the tank with live steam, and allow to cool and drain. Support the tank in a manner similar to the original installation as set forth in a foregoing paragraph. Attach an air pressure supply hose to one tank outlet and plug tightly all remaining outlets. Apply and maintain an internal air pressure of 5 lbs./sq. in. to the oil tank or 3-1/2 lbs./sq. in. to the fuel tank. The tank should withstand the specified pressure with no evidence of distortion. Thoroughly mix 1/2-cup of liquid castile soap with 1 cup of warm water, and with a soft-bristled brush, apply this solution to all seams, welded or riveted areas, and to all fitting edges. Carefully scrutinize the applied solution for evidence of soap bubble formation. If soap bubbles are found, indicating a leak, mark the location and continue the examination until the entire tank has been covered. If leaks are found, repair as outlined in the preceding paragraphs. If no leaks are found after a careful scrutiny of all possible areas, the tank is ready for use. Relieve the air pressure before removing the supporting jig. Wash all soap solution from the tank with a rag dipped in clean warm water.

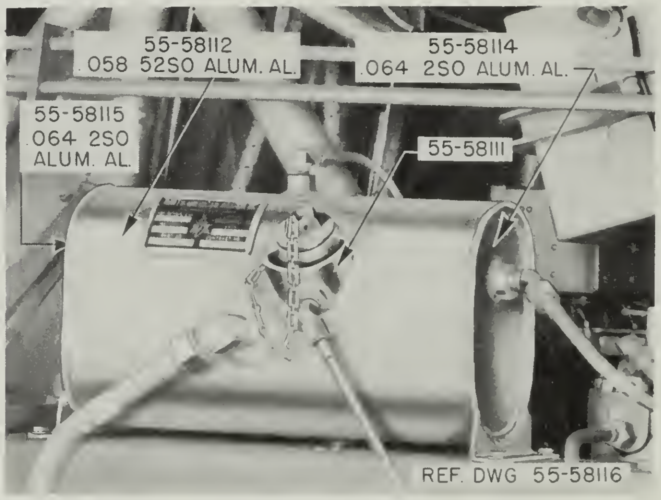


Figure 9—Hydraulic Oil Tank

11. REPAIR TOOLS AND MATERIALS.

To successfully accomplish the repairs outlined in this section, all or part of the following tools and materials are required:

TOOL OR MATERIAL	REMARKS
52SO ALUMINUM ALLOY INSERT MATERIAL	SIZE AND THICKNESS TO SUIT
52SO WELDING ROD	DIAMETER TO SUIT

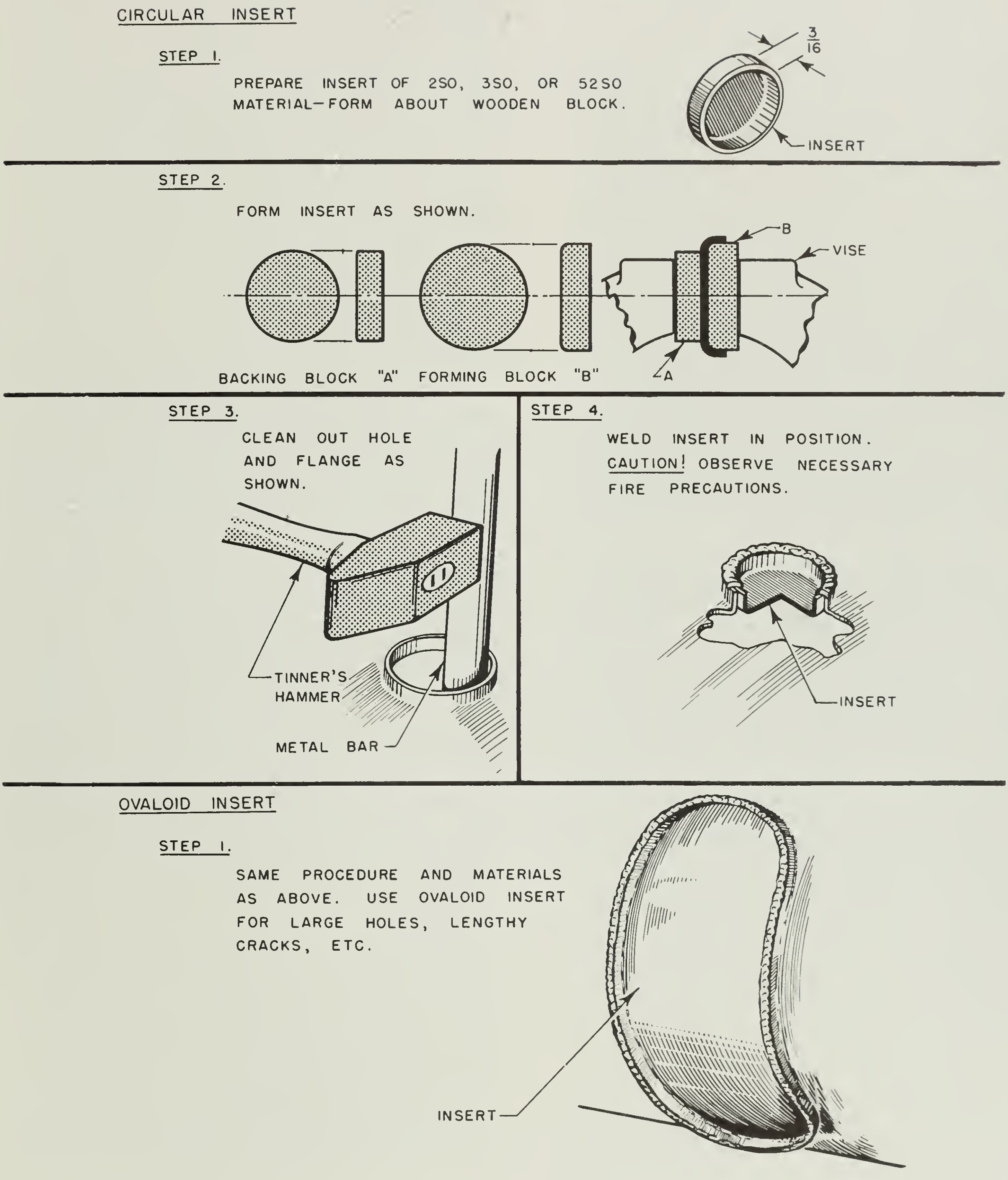


Figure 10—Forming and Welding Insert

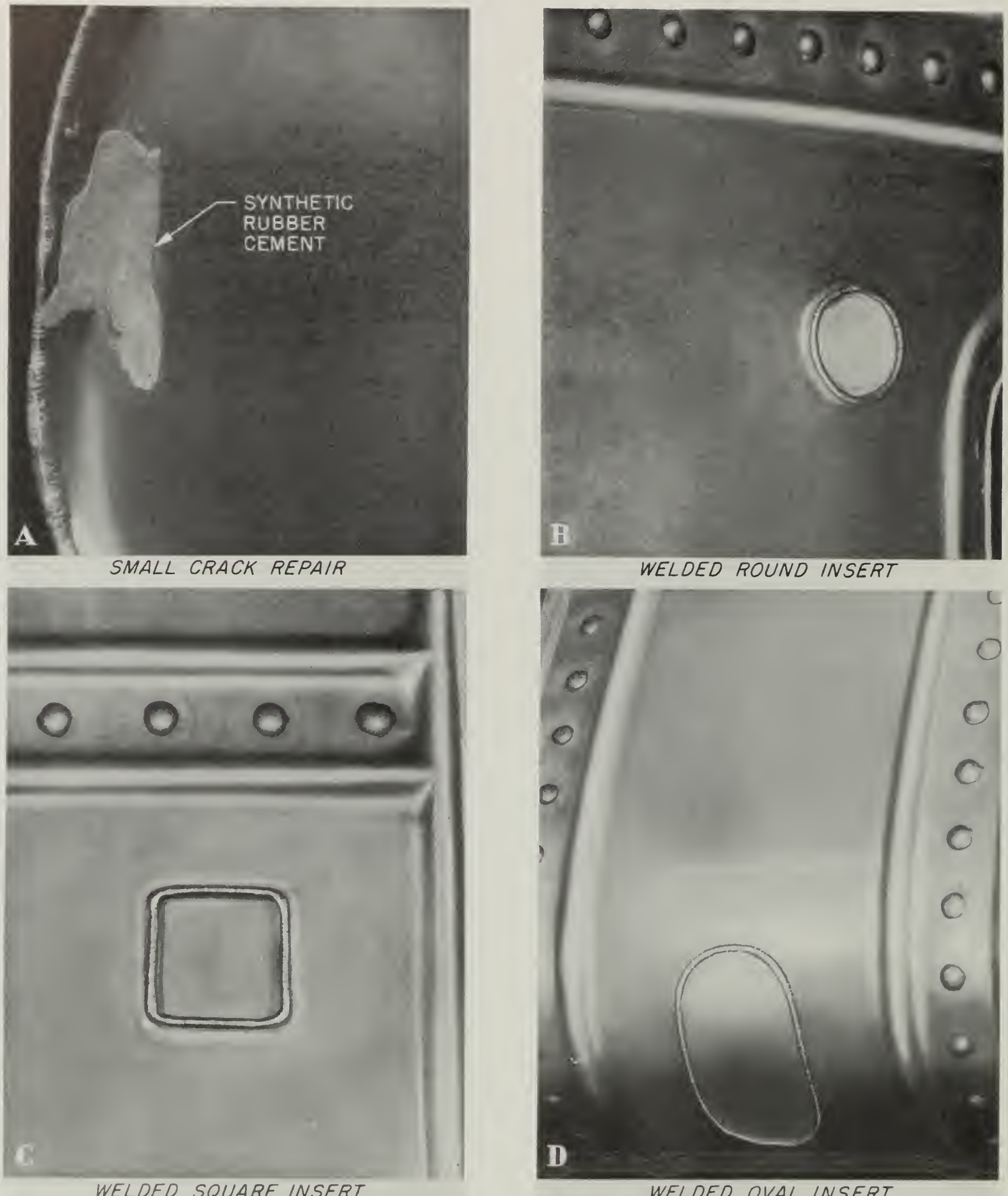


Figure 11—Completed Fuel and Oil Tank Repairs

TOOL OR MATERIAL	REMARKS
ALUMINUM ALLOY WELDING FLUX	PROCURE LOCALLY
HYDROGEN WELDING TORCH AND TANKS	OXYACETYLENE MAY BE SUBSTITUTED
SULPHURIC ACID	PROCURE LOCALLY
CARBON TETRACHLORIDE	PROCURE LOCALLY
LIQUID CASTILE SOAP	PROCURE LOCALLY
2-INCH SOFT BRUSH	PROCURE LOCALLY
SYNTHETIC RUBBER CEMENT, WATER AND GASOLINE RESISTING	DOLPHIN OR CAVALPRENE OR EQUIVALENT
RUBBER AIR HOSE AND SHUT-OFF VALVE	USED WITH TANK TESTING JIG
METAL BAR, HAMMER, AND WOODEN BLOCKS	USED TO FORM INSERTS.

OIL COOLERS

12. GENERAL.

A Type C-5 vertical oil cooler, on which is mounted a Type D-5 thermostatic relief valve, is located on the bottom of the engine accessory compartment and provides automatic oil temperature regulation for the engine. The oil cooler assembly is supported by means of a bracket extending from the top of the cooler and attached to the two lower engine mount tubes with rubber hose connections. *(See Figure 5.)* This Type C-5 oil cooler is manufactured by United Aircraft Products, Inc., Dayton, Ohio. Repair of these coolers may be necessitated by collapsed tubes, tube leaks, core surface leaks, surface leaks between the core and shell, assembly leaks through the silver solder bond, dents in the shell, and bullet holes in the shell or core.

13. OIL COOLER CONSTRUCTION.

The oil cooler is cylindrical in shape and has a diameter of 8 inches, a depth of 9-1/2 inches, a frontal area of 50.3 square inches, and a cooling surface of 29 square feet. Enclosed by a brass shell, the 9 inches long copper core tubes have .323-inch hexagonal ends, a .268-inch diameter, and a .006-inch wall thickness. By means of tin-lead solder, the hexagonal ends of the core tubes are held together, one tube to all its adjacent tubes, and the core assembly to the shell. When the cooler is installed, the oil seeps down between the round portion of the cooling copper tubes, the hexagonal ends of the tubes being sealed with solder. *(See Figure 12.)*

14. CLEANING OIL COOLERS.

Prior to making any repairs, remove the cooler and drain all oil in the cooler. Heat a quantity of carbon tetrachloride to the boiling point, pump the solution through the cooler, and strain the solution as it leaves the cooler. If bearing metal particles are found, the cooler should be replaced. After thoroughly flushing the cooler with the organic solvent, thoroughly clean the cooler with live steam for one-half hour. Allow the steam to pass downward into the internal passages, and position the cooler so that the condensed steam will drain freely from the bottom.

15. TESTING OIL COOLER FOR LEAKS.

If it is necessary to locate leaks in the cooler, seal all openings in the cooler except one, and to this opening apply an air pressure not to exceed 75 lbs./sq. in. Slowly submerge the cooler in warm water; and where bubbles appear, mark the location with a wire clip. Continue the examination until the entire cooler has been covered. Remove the cooler, disconnect the pressure line, and allow the cooler to drain.

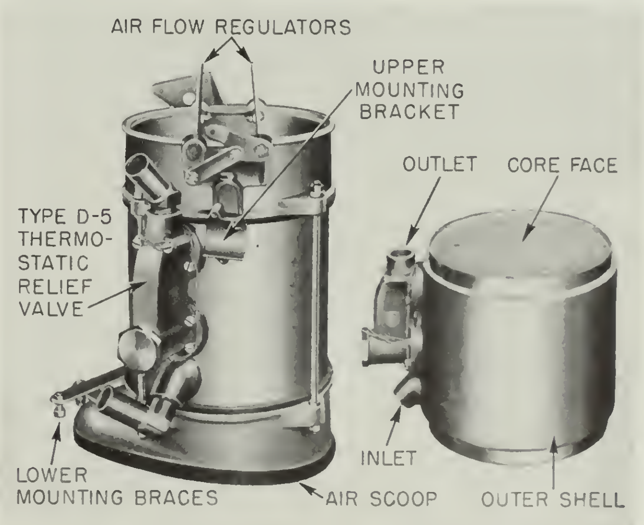


Figure 12—Oil Cooler Assembly

16. REMOVING SINGLE TUBES.

The best method for removing leaking or collapsed tubes is as follows: *(See Figure 13.)* Obtain at least two pieces of 5/16-inch hexagonal metal bar stock 4 inches long, and bevel the sharp edges from the ends. Heat these bars in a gas furnace; and while the irons are heating, push a 12-inch wire through the marked end of the defective tube in order to identify the corresponding tube ends. Flux both ends of the tube to be removed. With a small pair of tongs, remove each of the heated metal bars from the furnace, withdraw the identifying wire from the defective tube, and insert the tip of one iron into the tube end. As the wire is withdrawn from the opposite end, insert the tip of the other iron into that end. If the irons are hot enough, the solder bond should be broken immediately. When this occurs, gently tap one of the hexagonal bars until the tube is pushed out the opposite end. Remove the bars from both ends of the tube; and using a pair of pliers, pull the protruding end of the tube out through the core. Clean the adjacent tube surfaces. If any lumps of solder adhere to the adjacent tubes, shape the opening in the core with the hexagonal iron bar.

17. REPLACING TUBES.

Replacement tubes should conform to AAF Dwg. 065948-9. *(See Figure 13.)* Flux each end of the replacement tube and dip the hexagonal ends into molten tin-lead solder. Flux the openings in the core where the old tube has been removed and insert the new tube. With a pair of sharp-nosed pliers, re-form the hexagonal sides of the tubes adjacent to the new tube. After refluxing, use the standard soldering iron and melt tin-lead soldering wire over the face of the core with an oscillating motion of the iron.

18. REMOVING LARGE SECTIONS OF CORE.

The procedure outlined for the removal of single tubes should be used when removing one tube or several tubes. However, if a large section of the core must be removed, use an oxyacetylene flame on both ends of the core. Care must be taken to avoid the use of excessive heat when melting the solder bond from large sections of the core. Use only enough heat to melt the solder, then swiftly apply pressure to remove the core section.

19. LEAKS ON CORE SURFACE.

Clean the area around the core surface leak, and flux the surface with zinc chloride flux. Using a tin-lead solder wire and a hot iron, solder over the point of the leak with an oscillating motion of the iron. Leaks between the core and shell assembly may also be stopped by the application of solder.

20. LEAKS AROUND SHELL SEAMS.

To stop leaks around the silver-soldered seams of the shell assembly, thoroughly clean and flux the area. Using a soldering iron, melt a tin-lead soldering wire over the leak with an oscillating movement of the iron.

21. DENTS IN SHELL.

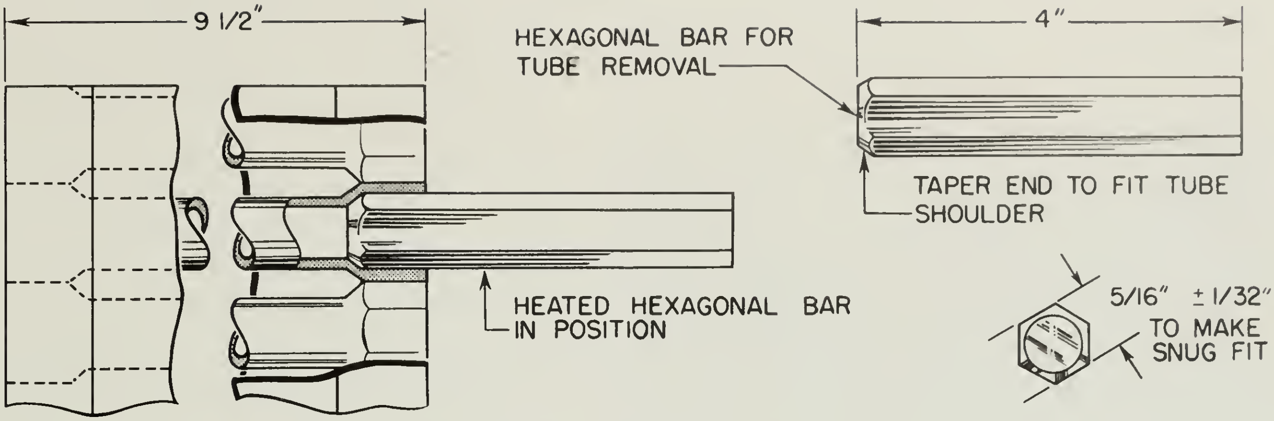
Large dents in the shell not indicating sharp radii may be corrected by the following procedure: Apply an air pressure of from 30 to 40 lbs./sq. in. to the inside of the cooler. Utilizing an acetylene flame, carefully heat the area affected by the dent. Under the air pressure from within, the material should return to its normal contour. Sharp dents sometimes may be corrected by soldering the end of the silver solder wire to the center of the dent. After this is done, gradually pull the shell back into position.

22. HOLES IN SHELL.

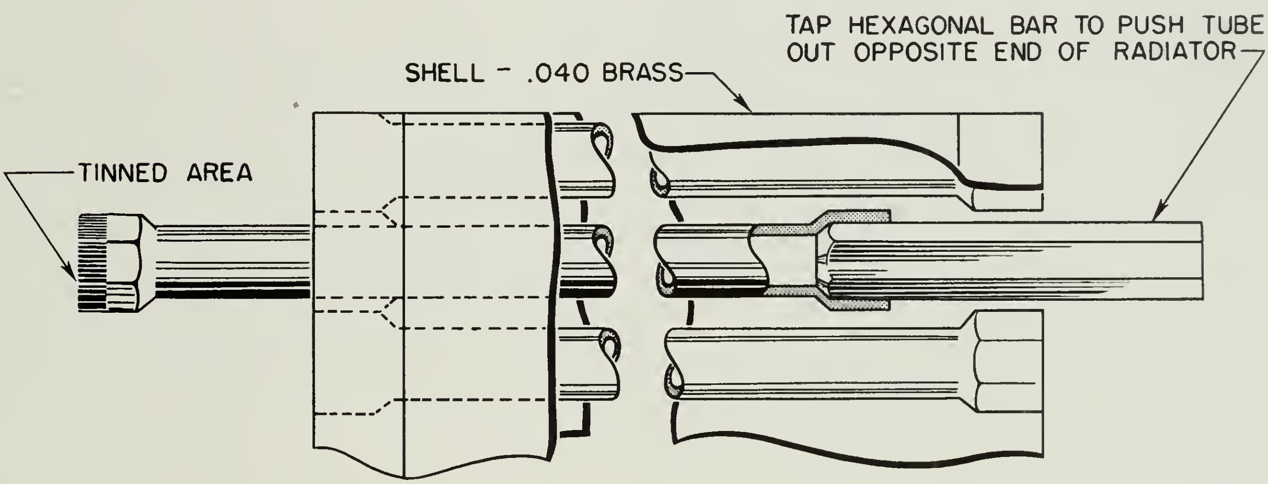
Small holes not exceeding 1/4-inch in diameter may be patched by utilizing a piece of .040 inch thick or .050 inch thick brass. Thoroughly clean the area with steel wool, and soft solder the patch over the hole. Large holes in the shell may be repaired by silver soldering, provided the core is properly protected from excessive heat by utilizing wet cloths. Holes in the inside shell are extremely difficult to repair, and the cooler should be replaced rather than repaired.

23. TESTING OIL COOLERS.

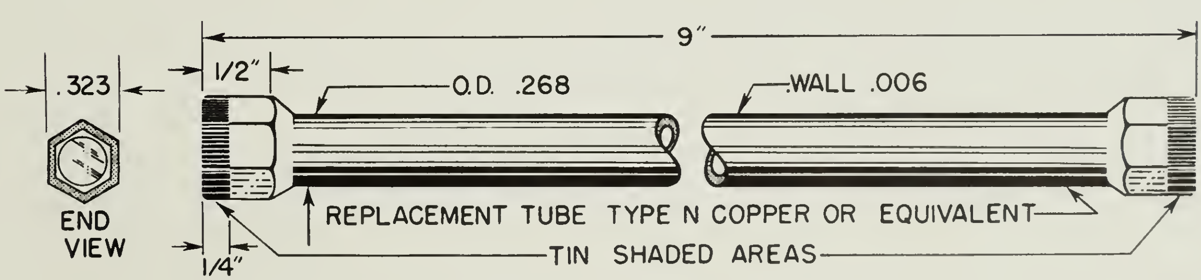
Upon completion of repairs, test the cooler as follows: Plug all outlets except one, and to this outlet apply an air pressure of 75 lbs./sq. in. Submerge the cooler in warm water and check for leaks. If a satisfactory repair is indicated, thoroughly flush the cooler with hot water and then apply live steam to the cooler interior for one-half hour. If the



MELTING SOLDER ON DEFECTIVE TUBE



TAPPING OUT DEFECTIVE TUBE



REPLACEMENT TUBE

Figure 13—Oil Cooler Tube Replacement

cooler is to be installed immediately, pump hot light engine oil through the cooler prior to installation. If the cooler is to be stored, prior to storage, heat a tank of SAE 20 engine oil to 121°C (250°F) and immerse the cooler in the oil. Vigorously agitate the cooler until the bubbling ceases. Remove the cooler, allow it to drain, and plug the fitting opening.

24. OIL COOLER REPAIR MATERIALS.

MATERIAL	REMARKS
FLUX, ZINC CHLORIDE	FOR BOTH SILVER AND TIN-LEAD SOLDER
WIRE, SILVER SOLDER	AAF SPEC. QQ-S-561 1/16-IN. DIAMETER
WIRE, TIN-LEAD SOLDER	AAF SPEC. QQ-S-571 1/8-IN. DIAMETER
TUBES, CORE REPLACEMENT	AAF PART NO. 065948-9
SHEET, BRASS	AAF SPEC. QQ-B-611 .025, .040, .050 IN. THICK, HALF HARD.

25. OIL COOLER REPAIR TOOLS.

Depending upon the extent of repairs to be undertaken, all or part of the following tools should be available.

TOOL	REMARKS
TORCH, OXYACETYLENE	
TANK, OXYGEN	
TANK, ACETYLENE	
REGULATOR, PRESSURE	
FURNACE, GAS-FIRED	SMALL
IRONS, SOLDERING	TWO LARGE ELECTRIC OR GAS-HEATED IRONS
BAR, 5/16-IN. HEX. METAL	FOR REMOVING TUBES, TWO BARS 4 IN. LONG REQUIRED.

SECTION 6
LANDING GEAR

1. GENERAL.

A fully retractable, hydraulically operated main landing gear and a nonretractable, full-pivoting tail wheel are provided. All components are of the single-leg, half-fork type and are equipped with air-oil-type shock struts. *(See Figure 1.)* Inasmuch as most repairs to the landing gear are accomplished by part replacement, procedures outlined in this Section deal mainly with the removal, repair, and reassembly of the tires, tubes, and wheels. Broken or cracked components of the landing gear assembly, or bent or chafed axles and pivot pins must be replaced. All special tools and materials mentioned are tabulated in the closing paragraphs.

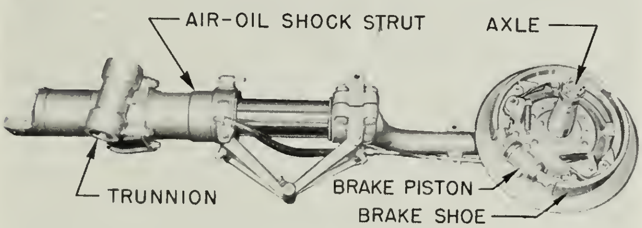


Figure 1—Main Landing Gear Shock Strut

Figure 2—Main Landing Gear

2. MAIN LANDING GEAR.

The main landing gear consists of two identical left-hand and right-hand components which retract inboard into the wing center-section just forward of the main spar on which they are supported. *(See Figure 2.)* On each component is mounted a one-piece, cast aluminum alloy, drop-center wheel assembly which is machined to permit installation of the steel drum liner, the bearing cups, and the 27-inch tire. The wheel access fairing is of stamped aluminum alloy and is easily removed to inflate the tire and inspect the bearings. Small raised inflation seams or deflection marks are provided on the periphery of each sidewall of the tire as an aid to proper inflation.

3. TAIL WHEEL.

In likeness to the main landing gear, the tail wheel assembly is also a single-leg, half-fork, full-cantilever structure. The tail wheel support assembly consists of an aluminum alloy casting and a chrome molybdenum steel forging knuckle assembly mounted on roller bearings within the rear portion of the support. *(See Figure 3.)* The assembly is attached to the fuselage aft section at three points. Two of these points are the main trunnions at the forward end of the cast aluminum alloy support to which the fork and the steering mechanism are mounted. The third point of attachment is located at the top of the pneudraulic shock strut. The axle and fork are of heat-treated chrome molybdenum steel. The fork is so heat treated as to make it the weakest item in the assembly and thereby prevent failure of more important units. Unlike the main landing gear, the tail wheel hub casting is made up of two parts as an aid to easy mounting of the tire and tube. *(See Figure 4.)* A 10-inch, smooth-contour tire is mounted on the tail wheel, the casing of which is also provided with small raised inflation seams or deflection marks on the periphery of each sidewall as an aid to proper inflation.

4. REMOVAL OF WHEEL ASSEMBLY.

To remove the main landing gear wheels, place an outer wing jacking stand under the outer wing jacking points and extend the jack until the wheel is free to rotate. The tank doors must be installed before jacking the airplane at the outer wing jacking points. An alternate method of jacking is accomplished by placing a small hydraulic jack in a main landing gear wheel jack cradle and placing the center of the jack under each landing gear strut at the jacking point provided. *(See Figure 5.)* Extend the jack until the wheel is free to rotate. This latter method

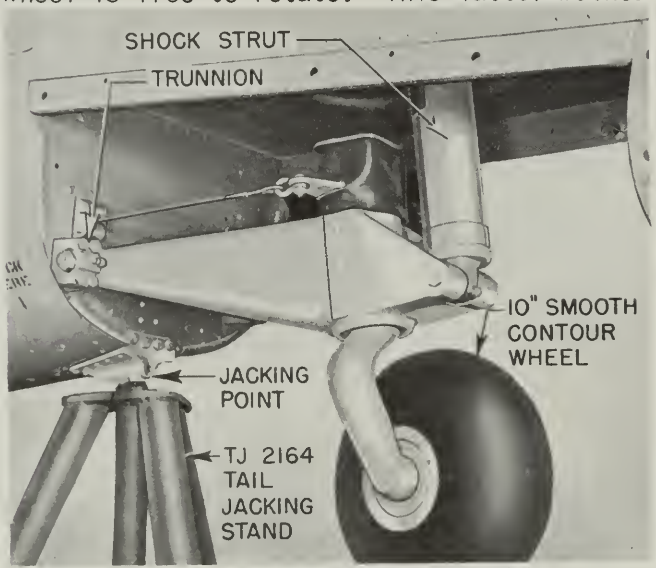


Figure 3—Tail Wheel Assembly

must necessarily be used if the wings are not attached. Remove the dzus-fastened access fairing plate. Remove the hub nut cap retaining spring and the hub nut cap. Extract the cotter key and unscrew the large hex nut on the axle. *(See Figure 6.)* To remove the tail wheel, place a jack under the tail jacking point located at station 215-5/8 and extend the jack until the wheel is free to rotate. Remove the screw-fastened wheel fairing plate; extract the cotter key and remove the retaining nut and washer from the axle. *(See Figure 7.)*

5. REMOVAL AND INSPECTION OF TIRE AND TUBE.

Remove the tire from the wheel with any standard or improvised tire tools which are free from sharp edges. Care must be taken not to injure tire, tube, or wheel with these

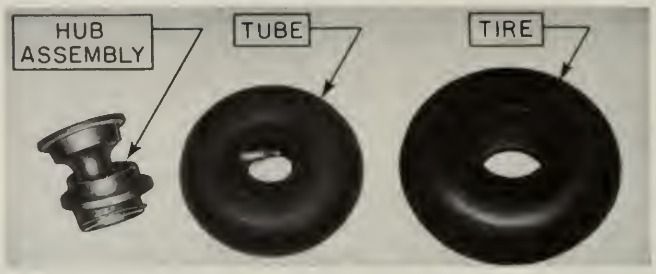


Figure 4—Tail Wheel Components

tools. Inspect inside and outside of tire casing for damages. Classify the damage as repairable or not repairable, and treat accordingly. External defects of the casing or damage which does not affect the fabric may be repaired. All damaged casings shall be repaired except those having one or more of the following defects: internal breaks, cuts or tread wear which exposes the fabric carcass; diagonal or X breaks; injuries to the bead or bead area, except in cases where the chaffer fabric is damaged or loose, provided the damage is not caused by working of the beads; evidence of body ply separation or severe sidewall flexing; injuries through at least 50 percent of the cord body and larger than 2 inches; sidewall injuries requiring patch repairs; two sidewall injuries

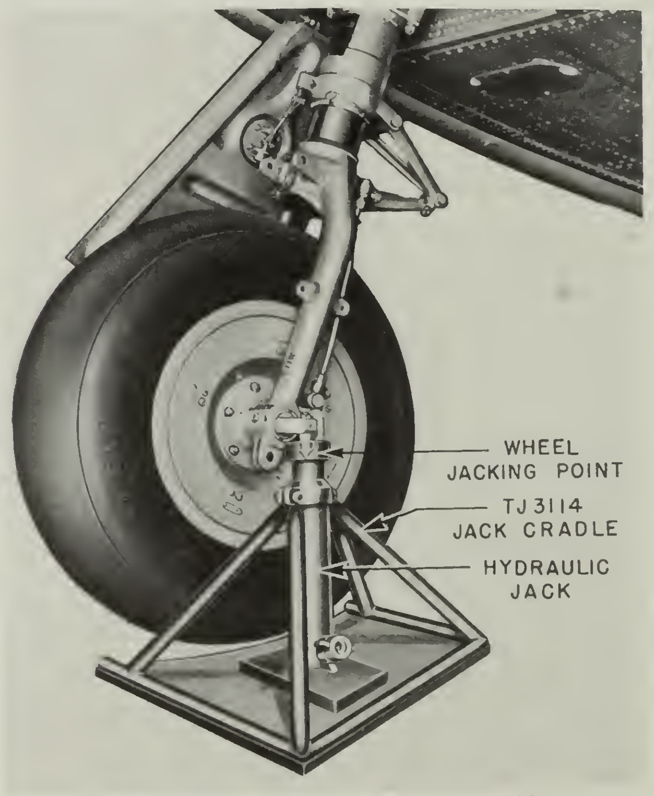


Figure 5—Hydraulic Jack Cradle

within 60 degrees of each other; and injuries of any kind that cannot be repaired in a satisfactory manner. Worn casings retaining more than 25 percent of their estimated service life and having no irreparable injuries shall be treaded except when the casing has been treaded twice or where the tread is unserviceable because of cactus spines. Inspect the tube for the following defects: physical damage to the valve or faulty attachment to the tube; tube body defects, or evidence of wrinkles or creases; tube body thin spots, cuts, punctures, chafing, or damaged areas resulting from casing breaks. To expedite repair, mark all injuries as they are found.

6. WHEEL INSPECTION AND REPAIR.

After the tire and tube have been removed from the wheel, inspect the rim for evidence of corrosion, cracks resulting from careless use of tire tools or impact of any object, and worn places in the protective coating of paint on the rim. If the wheel becomes damaged to the point where excessive distortion is noted or where cracks appear in the casting, no attempt should be made to repair the wheel or rim. Portions of the wheel where the original finish has become worn or has peeled off should be retouched with lacquer. One or two coats should be applied, depending upon the condition of the old paint. If it is necessary to refinish the surface completely, one coat of primer followed by two coats of lacquer should be applied. It is essential that the surface be thoroughly cleaned before the application of the primer. The old finish should be removed with acetone.

7. NEW CASING REPAIR.

New casings having defects of a minor or major nature may be repaired before use. Repair all defects of a new casing in the same manner as set forth in the following paragraphs for used casings. The following defects appearing in new casings shall be considered negligible: lettering illegible on one side only; portions of deflection markers missing provided that at least 75 percent of the marker is visible on each side of the casing; rounded edges of tread design due to lack of pressure when curing, when the radius does not exceed one half of the nonskid depth; depressions in the tread or sidewalls caused by foreign material or air traps when the presence of such depressions does not affect the serviceability of the casing; inside indentations, caused by foreign substances, when such indentations are smooth and shallow; spread cords when the space between cords does not exceed the width of one cord and when the cord count over a 15-degree arc of the casing is not reduced more than 15 percent.

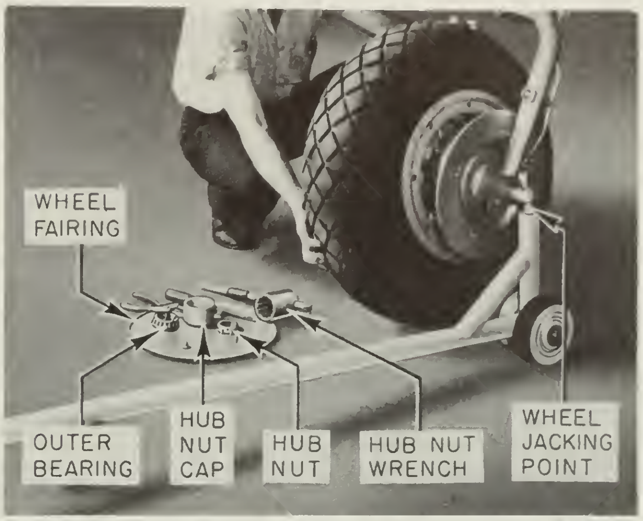


Figure 6—Removing Main Landing Gear Wheel

8. CASING REPAIRS.

All repairs to casings shall be made with standard commercial type uncured or semicured cord fabric. The number of cords and the quality shall be the same as those in the original casing. If any of the repairable defects or damages listed in a foregoing paragraph are present, repair as follows: Remove all weakened or weathered cords and roughen all skived tread surfaces. Thoroughly clean and buff all cord surfaces to which new material is to be applied and coat all exposed cords with cement immediately after they are exposed. Repair injuries through the tread or sidewall only and those smaller than 1/2-inch extending into not more than 25 percent of the cord body plies by filling the entire hole with gum and applying a small piece of sheeted gum stock over the hole on the inside cord ply. Reinforce with an inside patch all injuries larger than 1/2-inch or extending through more than 25 percent of the cord body plies. Clean, cement, and replace all chaffers damaged by tire tools in removing the casing from the rim. Cure all repairs by any suitable means. If original type molds are used, register the tread design on the casing with the tread design in the mold. Regulate the time and temperature of the cure so that it will be appropriate for the rubber stock being cured. Tread cuts which do not penetrate the fabric are to be cleaned, filled with commercial tire cut filler, and cemented in place. If sidewall blisters are present that can be cleaned and repaired with no resultant damage to the fabric carcass, or if the rubber fairing immediately above the rim flange separates from the fabric carcass with no apparent injury to the carcass, the damaged areas may be repaired by cleaning with wash thinner or gasoline, buffing the areas to be cemented, and cementing separated areas with airship rubber cement. Allow cement to become tacky, and press rubber sidewall firmly against the fabric carcass. Some means should be provided to maintain a constant pressure on the repaired area until cement has become thoroughly dry.

Figure 7—Removing Tail Wheel

9. TREADING.

All treading materials and workmanship shall be in accordance with high-grade commercial practice covering this type of repair work. Repair all damaged casings except those having one or more of the defects listed in a following paragraph. Normally, the treading operation shall be confined to the entire portion of the wearing surface of the casing that is between and including the inflation deflection markers. Replace the deflection markers. When treading a casing having the tread or breaker plies separated from the carcass or having deep cuts in the tread, include as much of the sidewall as is necessary to produce a serviceable casing. Remove the old tread and portions of the sidewall where necessary and thoroughly roughen the working surface. Cover the working surface with vulcanizing cement and apply the new stock. Care must be taken when rolling down the new stock so as to exclude all air from between the new stock and the original carcass. Cure the treaded casing in full circle or factory-type molds, using appropriate time and temperature for the rubber stock being cured. Clean and paint the treaded casing after removing it from the mold.

10. MARKINGS.

All repaired or treaded casings shall be properly marked upon completion of repair. Identify a repaired casing by painting or rubber stamping an arrow and the letter R on the repaired area. Use suitable red or yellow paint or enamel for this purpose. Burn the brand REP on each repaired casing immediately following the original serial number. Apply the above brand each time a casing is repaired, in addition to all former brands. Burn the brand TR on each treaded casing immediately following the original serial number. If the original serial number was removed in the treading operation, replace it by some approved method. Apply the TR brand each time a casing is treaded, in addition to all former brands. All casings repaired or treaded by other than the original manufacturer must have the contractor's name or trademark burned upon them.

11. TUBE REPAIR.

Thin spots, cuts, and punctures may be repaired by patching or vulcanizing by any approved method. The patch size and the number of patches should be limited, as the weight of the patch or patches may cause tube unbalance. If large holes or extensive thin or chafed areas are present, the tube must be discarded.

12. TIRE AND TUBE BALANCE.

In order to prevent dynamic unbalance of the landing gear tires at high speeds, it is necessary to balance the tires and tubes statically. The maximum static unbalance for a 27-inch smooth-contour tire as set forth in the AAF Spec. AN-C-5 is 12 inch-ounces. Inasmuch as attaching balance weights on the outside periphery of the tire is difficult, balancing methods set forth in the following paragraphs will designate all weights at the rim line. Twelve inch-ounces on the outside periphery of the casing is equal to approximately 2-1/2 ounces at the rim line. Hence, all main landing gear tires should not exceed a maximum static unbalance point of 2-1/2 ounces at the rim line. The tail wheel need not be balanced.

13. TUBE BALANCING.

In order to balance statically a patched tube, a balance wheel is needed. *(See Figure 8.)* One may be fabricated from wood or from a discarded wheel. The balance wheel must be statically balanced before any attempt is made to balance the tire or tube. Inflate the tube until the true shape is reached but no stretching occurs. Mount the tube on the balance wheel and allow the wheel to rotate. The heavy portion of the tube will seek the lowest point on the wheel. Mark this point, rotate the wheel 90 degrees, and add a balance weight at the rim line. Experiment with different weight values until the accurate amount of weight required to balance the tube is found. If this weight exceeds 2-1/2 ounces, patches may be added 180 degrees from the heavy spot for a counterbalance. Apply all balance patches with the tube fully inflated. When the weight required is less that that mentioned, mark the light area of the tube to facilitate wheel balancing upon insertion of the tube into the casing.

14. TIRE BALANCING.

Mount tire on balancing wheel and determine the balance weight as directed for tube balancing in the preceding paragraph. *(See Figure 8.)* As soon as the correct amount of balance weight is found, mark the heavy section. If the static unbalance exceeds 2-1/2 ounces at the rim line, remove the tire from the balance wheel and spread the beading apart with wooden blocks directly opposite the heavy section. The inside area of the casing to which balancing dough is to be applied is indicated by a dotted line. *(See Figure 8.)* Clean the inside of the casing for an area of at least 1-1/2 square feet, using a wire brush and a rag slightly moistened with wash thinner. Apply one coat of Goodyear rubber cement No. 865 with a 2-inch stiff brush. *(See Figure 9.)* Allow to dry for

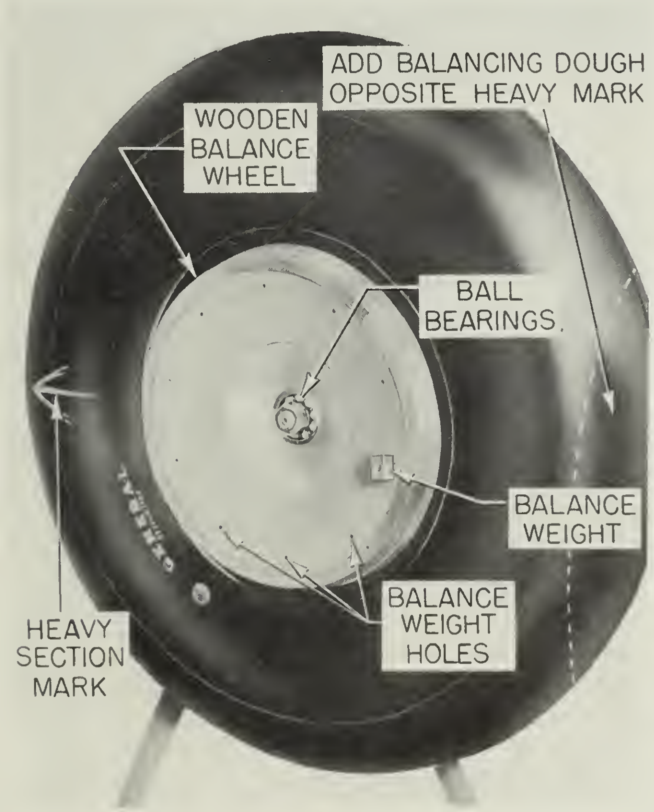


Figure 8—Balance Wheel

Figure 9—Applying Cement

Figure 10—Applying Balancing Dough

Figure 11—Applying Tire Talc to Balancing Dough

30 minutes and apply Goodyear No. 865C balancing dough cement with a metal spreader. *(See Figure 10.)* Allow to dry for 30 minutes. The amount of dough applied will depend upon the weight required to balance the tire. Recheck the tire balance on the balance wheel; add or remove balancing dough until balancing weight is within the minimum of 2-1/2 ounces. After the balancing dough is dry, apply powdered tire talc to the balancing dough. *(See Figure 11.)* This prevents the tube from sticking to the balancing dough. The tire is now ready for installation.

15. ASSEMBLING TIRE AND TUBE ON WHEEL.

Lay the wheel on a flat smooth surface with the drop-center rim portion toward the top. Utilizing standard tire tools and a rubber mallet, mount one bead of the casing on the rim. *(See Figure 16.)* Before inserting the tube, dust the entire tube surface with powdered tire talc. This may be best accomplished by placing a quantity of tire talc in a shallow box, laying the tube on the bottom, and applying the tire talc with a cheesecloth. *(See Figure 16, Detail B.)* The tube may also be dusted when fully inflated if a dusting box is not available. *(See Figure 12.)* Tie a 2-foot cord securely onto the valve stem, or screw on a valve extension. *(See Figure 13.)* Insert the tube into the casing, carefully aligning the manufacturer's tire and tube heavy and light section marks. These appear in the form of a red dot or mark on the side of the tube and the casing. *(See Figure 14.)* Alignment of these two red dots is essential to wheel balance upon installation of the tire and the tube on the wheel. Place the valve stem opposite the valve hole. Thread the valve stem string or extension through the valve stem hole in the rim, screw on the valve locknut a few turns, and add just enough air to shape the tube properly. If too much air is added, difficulty will be encountered when mounting the remaining bead of the casing. Mount the second bead of the casing, starting at the valve. The use of tire-holding clips will expedite the mounting of the second bead

Figure 12—Applying Tire Talc to Inflated Tube

of the casing. *(See Figure 15.)* After the casing is properly mounted, add air slowly until the casing beads seat. If the beads of the casing do not seat properly, deflate the tube, realign the casing, and reinflate. Remove the valve core, and fully deflate the tube to relieve the pressure on any folds or buckles and to permit the tube to assume its proper contour within the casing. Be sure both casing beads

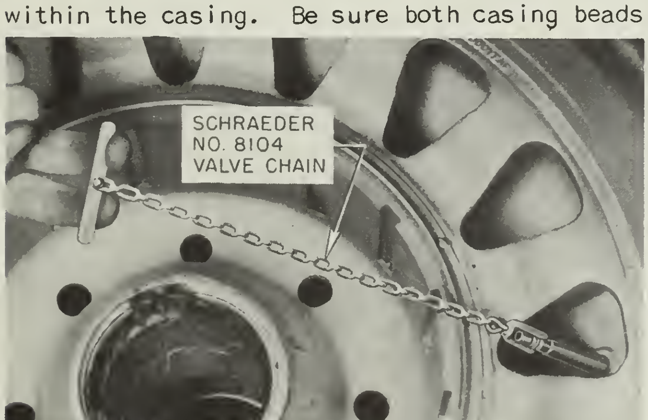


Figure 13—Valve Extension

remain properly seated on the rim. Replace the valve core, inflate the tire to the required pressure, and tighten the valve locknut. Inflate the tire until the deflection marks on the side of the tire just contact the ground line when the airplane is in a three-point position on hard level ground. Under normal conditions, this inflation will be equivalent to approximately 30 lbs./sq.in.

16. MOUNTING TAIL WHEEL TIRE AND TUBE.

Inasmuch as the hub of the tail wheel is of a two-piece construction, the mounting of the tire and tube of the tail wheel is not similar to that of the main gear. Apply tire talc to the tube by placing the tube in a dusting box and applying tire talc with a cheesecloth. *(See Figure 18.)* Insert the tube into the casing

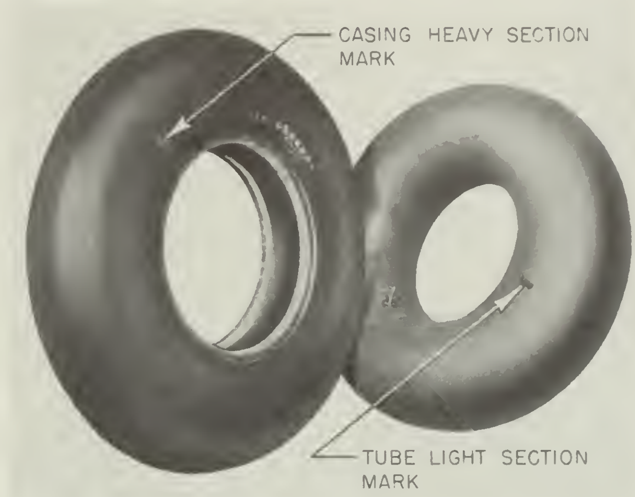


Figure 14—Tire and Tube Markings

and partially inflate. *(See Figure 17, Details A and B.)* Remove the demountable rim portion of the hub and slip the tire and tube over the remaining portion. *(See Figure 17, Details C and D.)* Thread the valve stem through the valve stem hole and place the demountable portion of the hub in position. Place the retaining spring in position and inflate the tube. *(See Figure 17, Details E and F.)*

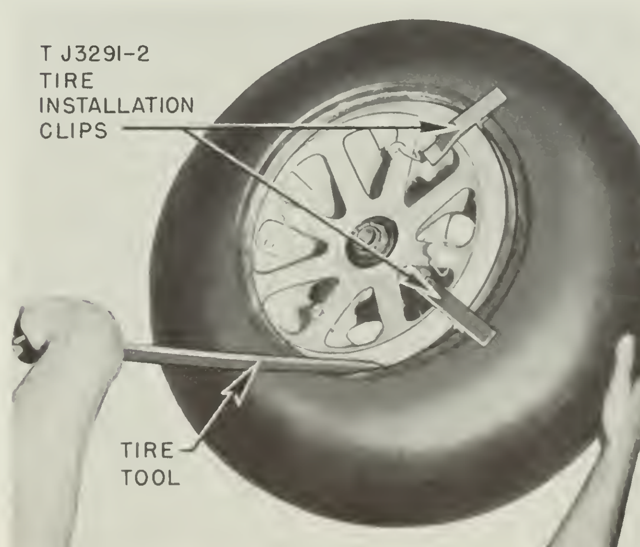


Figure 15—Special Tire Tools

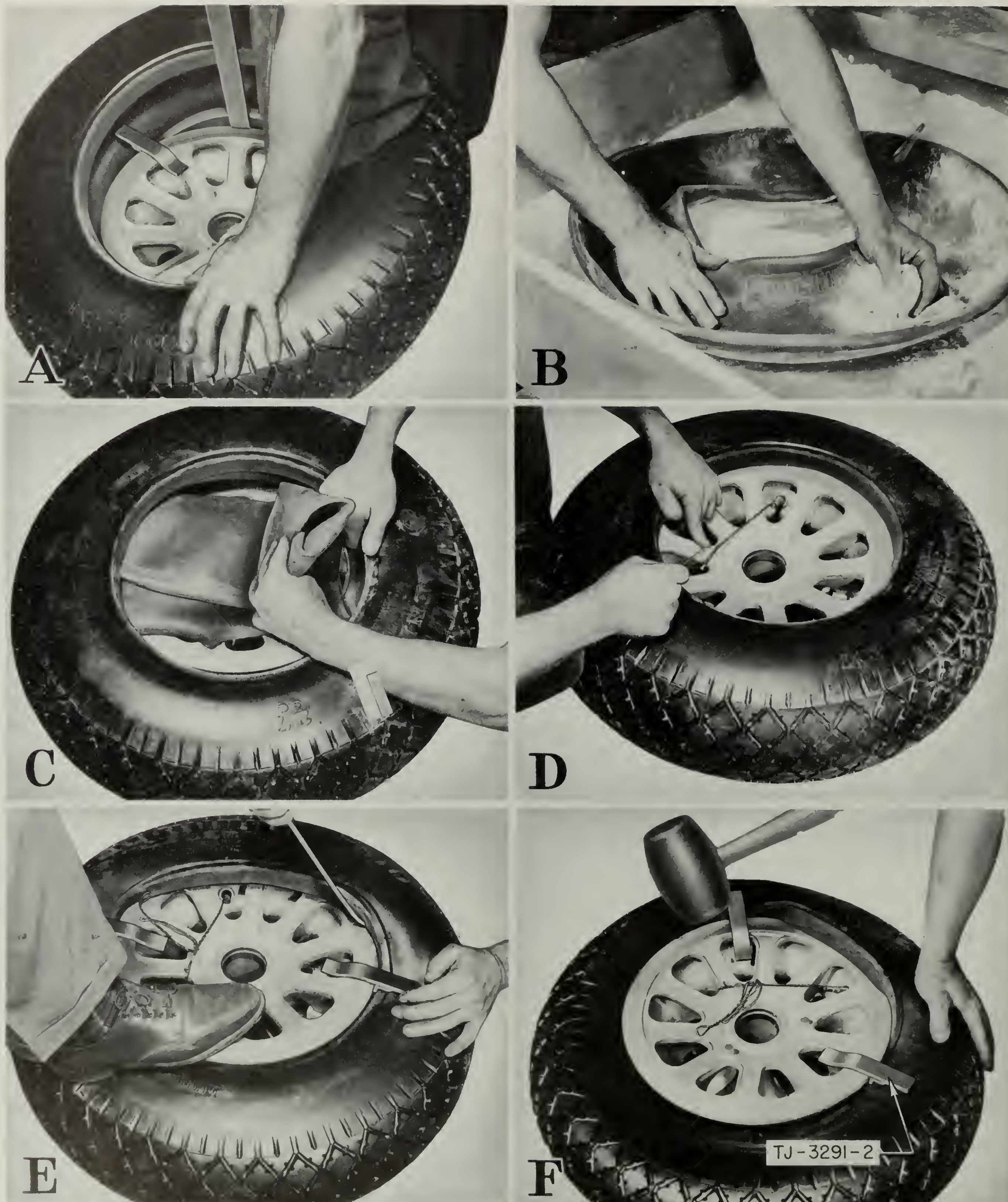


Figure 16—Mounting Main Gear Tire and Tube

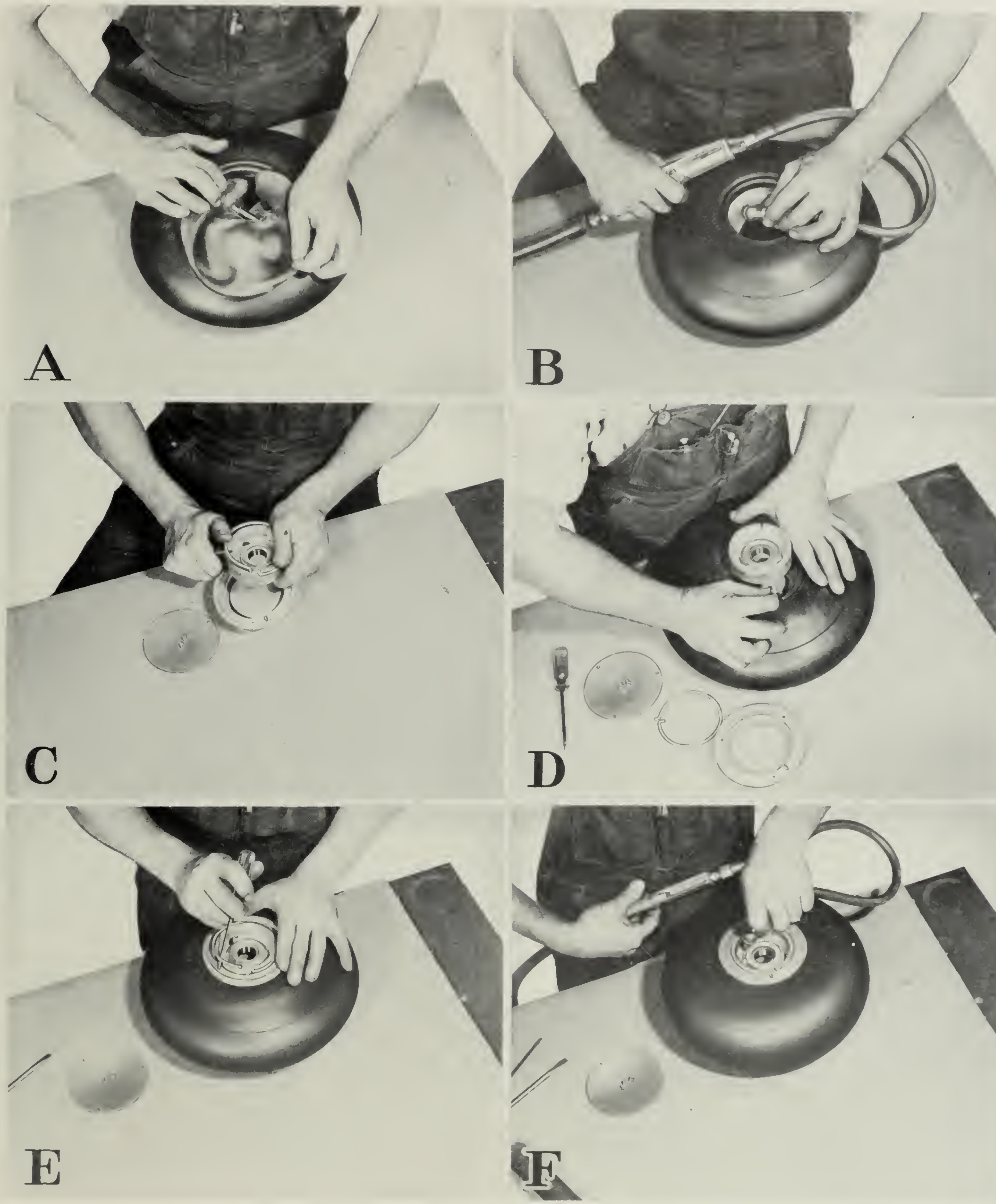


Figure 17—Mounting Tail Wheel Tire and Tube

Figure 18—Applying Tire Talc to Tail Wheel Tube

17. TOOLS.

Jacking stands and special tools are obtainable to expedite wheel jacking, removal, and reassembly. All special tools listed are manufactured by North American Aviation, Inc., for use with this series of airplanes. To remove, repair, and reinstall landing gear wheels and tires properly, all or part of the following tools are required:

TOOLS	REMARKS
CRADLE - JACK	PART NO. TJ 3114
STAND - WING JACKING	PART NO. TJ 4612
STAND - TAIL JACKING	PART NO. TJ 2164
WRENCH - MAIN AND AUXILIARY LANDING GEAR HUB NUT	PART NO. TJ 4429
TOOLS, TIRE	1-1/2 INCHES BY 18 INCHES, OR LARGER
MALLETS, RUBBER	LARGE OR MEDIUM
EXTENSION, VALVE, FLEXIBLE	AT LEAST 6 INCHES LONG
JACK, SMALL, HYDRAULIC	USE UNDER WHEEL JACKING POINT

18. MATERIALS.

The following materials are needed for repairs outlined in this Section:

MATERIALS	REMARKS
CEMENT, AIRSHIP RUBBER	SPEC. 20-37 CLASS 04-B
TALC, POWDERED, TIRE	SPEC. 4-33 CLASS 29
LACQUER, CELLULOSE NITRATE	SPEC. 3-158
PRIMER, METAL ZINC CHROMATE	SPEC. 14080
ACETONE	SPEC. 0-A-51A
FILLER, RUBBER TIRE CUT	COMMERCIAL TYPE, PROCURE LOCALLY

SECTION 7

CONTROL CABLES

1. GENERAL

The control cables are fabricated mostly of extra-flexible preformed corrosion-resistant steel; however, tinned steel cable is also used interchangeably. Cables vary from 1/16 to 5/32-inch in diameter. Cables of 1/8-inch and larger are composed of seven strands of 19 wires each. Cables 1/16 and 3/32-inch in diameter are composed of seven strands of seven wires each. The main control cable terminals are of the die-swaged friction type. The remaining cables are either sweat-soldered into tinned terminals or woven spliced. For information pertinent to the type cables and cable terminals used in original fabrication see Figures 1 and 2. It is to be noted that these figures list only the terminals directly swaged, soldered, or spliced to the cable. Miscellaneous fittings subsequently screwed to the end terminals are not listed.

2. NEGLIGIBLE DAMAGE

During overhaul periods, or more frequently if circumstances permit, test the control cables for broken wires by passing a cloth along the length of the cable. Broken wires will be indicated where the cloth is snagged. Broken wires should not be permitted in the cables of the trim tab control system, nor of the main control system where such breakages occur in that length of a cable normally passing over a pulley or through a fairlead. A maximum of seven broken wires along a straight uninterrupted length may be permitted without replacement of the cable. Broken wires, when found, should be cut off short and served or soldered to the cable. It is recommended that records designating the status of the damaged cables be maintained as an aid in determining the future disposition should additional damage occur.

3. CABLE REPLACEMENT

Wherever possible, duplicate spare cables should be utilized for replacements. See Figures 1 and 2 for the part numbers of all control cables.

4. CABLE FABRICATION

If spare cables are not available, exact duplicates of damaged cables should be prepared in accordance with the data contained in Figures 1 and 2. However, if facilities and supplies are limited and immediate replacement is imperative, replacements sometimes may be prepared in the customary manner, using thimbles, bushings, and turnbuckles in place of original terminals. When this is done, cables having a diameter of 3/32-inch or over may be woven spliced by means of the five-tuck method, and cables less than 3/32-inch in diameter may be wrap-soldered. The following chart specifies the particular terminals recommended for use with an improvised woven spliced and wrap-soldered cable only.

CABLE SIZE	THIMBLE	BUSHING	TURNBUCKLE ASSEM. (INCLUDING FORKS)
1/16	AN100-3	AN111-3	AN150-16S
3/32	AN100-3	AN111-3	AN150-16S
1/8	AN100-4	AN111-4	AN150-21S
5/32	AN100-5	AN111-5	AN150-32S

5. MATERIAL SPECIFICATIONS.

The following are the U.S. Army Air Forces Specification Numbers (except as noted) for the materials quoted in the following processes.

CABLE, CORR. RESIST. STEEL	AN-RR-C-48
CABLE, TINNED STEEL	AN-RR-C-43
CORD, SERVING	V-T-291
COROL NO. 95 SOLDER	QQ-S-571
WIRE, WRAPPING	AN-QQ-S-435

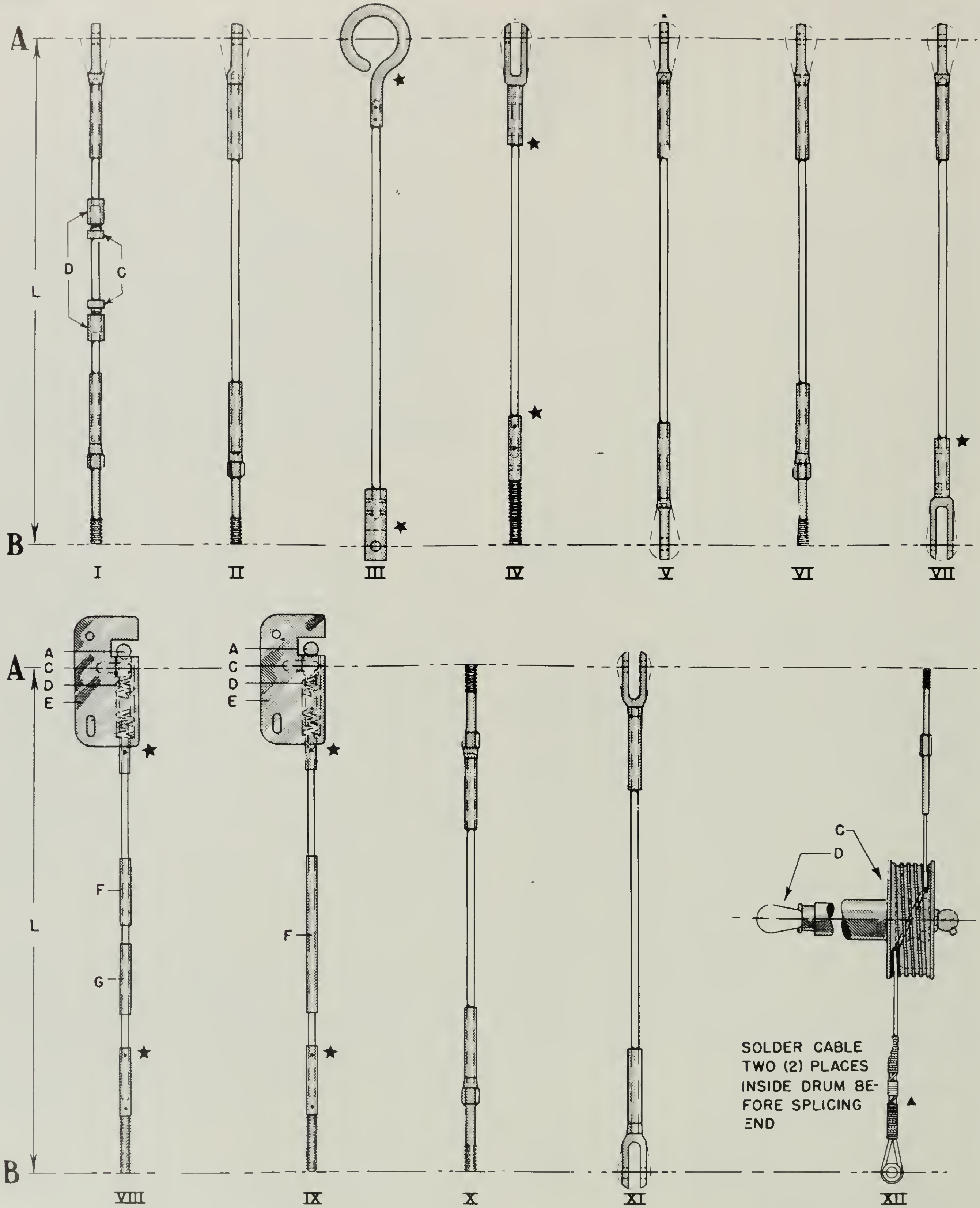
7

6. CUTTING CABLES

Cables may be cut by any satisfactory method provided measures are taken to prevent strands from fraying. Fraying may be prevented by serving the end with a short length of cord or wire or by sweating solder into the cable prior to cutting. The approximate length of the cable assembly may be determined from Figures 1 and 2.

7. RUST PREVENTION

If the cables are made from tinned steel, a rust preventive oil such as Simoniz Corol No. 95 or equivalent should be utilized to coat the cable. Dip the cable into a tank of Corol at 77° ± 5°C (170° ± 10°F) for 1/2-minute. Remove the cable and allow it to drain for 1 minute.



ALL FITTINGS ARE SWAGED EXCEPT AS NOTED

▲ WRAP SOLDERED SPLICE
★ SOLDERED FITTING

Figure 1—Types of Cables

CABLE LOCATION	NO. REQD.	N.A. DWG. NO.	TYPE	FITTING A	CABLE DIA.	FITTING B	LENGTH IN INCHES	ADDITIONAL FITTINGS AS NOTED
PARKING BRAKE	1	77-33450	VII	NAF310621B-2	1/16	AC066135	25-15/16	
REAR BRAKE - REAR RUDDER PEDAL TO MASTER BRAKE CYLINDER	2	66-33476	VI	NAF310621B-5	5/32	NAF310621-S-32D-5RH	72-11/32	
FRONT BRAKE - FRONT RUDDER PEDAL TO MASTER BRAKE CYLINDER	2	66-33477	V	NAF310621B-5	5/32	NAF310621B-5	12-5/8	
LANDING GEAR POSITION INDICATOR	1	36-33562-2	IV	AC066135	1/16	25-33526	53	
LANDING GEAR POSITION INDICATOR	1	36-33562-3	IV	AC066135	1/16	25-33526	88-3/8	
TAIL WHEEL CONTROL	2	66-34012	VI	NAF310621B-5	5/32	NAF310621-S32D-5LH	33-5/32	
MANUAL STARTER ENGAGING CONTROL	1	77-45041	III	77-45039	1/16	77-45042	25-1/16	
ELEVATOR FRONT LOWER	1	66-52217	VI	NAF310621B-4	1/8	NAF310621-S21D-4RH	215-7/32	
ELEVATOR FRONT UPPER	1	66-52218	VI	NAF310621B-4	1/8	NAF310621-S21D-4RH	173-11/32	
ELEVATOR REAR	2	77-52219	VI	NAF310621B-4	1/8	NAF310621-S21D-4LH	45-19/32	
AILERON CONTROL FRONT - WING C.S.	2	66-52310	II	B1277-4	1/8	NAF310621-S21D-4RH	92-9/16	
AILERON CONTROL REAR - WING C.S.	2	66-52311	II	B1277-4	1/8	NAF310621-S21D-4RH	90-5/32	
AILERON CONTROL FRONT - OUTER WING	2	66-52312	VI	NAF310621B-4	1/8	NAF310621-S21D-4LH	61-21/32	
AILERON CONTROL REAR - OUTER WING	2	66-52313	VI	NAF310621B-4	1/8	NAF310621-S21D-4LH	69-15/32	
RUDDER CONTROL FRONT LH	1	66-52459	V	NAF310621B-4	1/8	NAF310621B-4	165-1/8	
RUDDER CONTROL FRONT RH	1	77-52459	V	NAF310621B-4	1/8	NAF310621B-4	165-5/16	
RUDDER CONTROL REAR LH	1	66-52460	VI	NAF310621B-4	1/8	NAF310621-S21D-4LH	63-27/32	
RUDDER CONTROL REAR RH	1	77-52460	VI	NAF310621B-4	1/8	NAF310621-S21D-4LH	63-5/32	
RUDDER - BALANCE	1	66-52461	I	NAF310621B-3	3/32	NAF310621-S16D-3LH	111-9/16	23-52453(C) 23-52464(D)
ELEVATOR TRIM REAR	2	66-52527	XII	B1284-L	1/16	AN111-3	231-1/8	19-52510(C) 19-52505(D)
ELEVATOR TRIM FRONT	1	66-52528	XI	NAF310621C-2	1/16	NAF310621C-2	306-15/16	
RUDDER TRIM - FRONT	1	66-52537	X	B1284-R	1/16	B1284-R	345-3/8	
RUDDER TRIM - REAR	1	66-52538	XII	B1284-L	1/16	AN111-3	194-5/8	19-52510(C) 19-52505(D)
FLAP POSITION INDICATOR	1	55-52629	IV	AC066135	1/16	25-33526	77-7/16	
INSTRUMENT FLYING HOOD RELEASE LATCH - FRONT	1	58-73012	IX	55-73015	1/16	25-33526	48-3/4	52-73010(C) 55-73011(D) 52-73016(E) 55-73005(F)
INSTRUMENT FLYING HOOD RELEASE LATCH - REAR	1	59-73034	VIII	55-73015	1/16	25-33526	85-3/8	52-73010(C) 55-73011(D) 52-73016(E) 59-73005(F) 58-73035(G)

Figure 2—Cable Data

Wipe off all excess oil. It is to be noted that corrosion-resistant steel cable does not require this treatment for rust prevention.

8. SWAGED TERMINALS

Swaged terminals should be utilized in fabrication only if they were originally employed. The specific terminals noted on Figures 1 and 2 must be used in their original function. After preparing the necessary cable length with allowance made for the fitting elongation under swaging and proof loading, coat the end of the cable with No. 10 SAE lubricating oil. Insert the cable into the terminal approximately 1 inch, bend the cable toward the terminal, straighten the cable back to normal position, then push the cable end entirely into the terminal. Apply a drop of light oil to the terminal and insert the terminal about 1/4-inch into the die when the die is moving at a slow speed. Then insert the fitting entirely into the die and adjust the die for regular speed *(see Figure 3)*. If swaged terminals are used on both cable ends, prior to the second swaging operation,

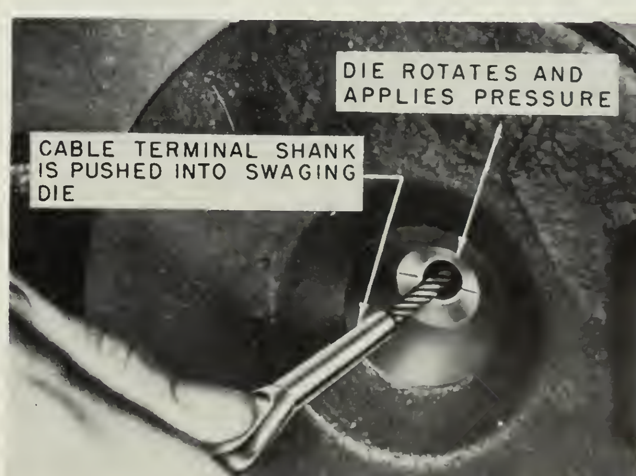


Figure 3—Cable Terminal Swaging

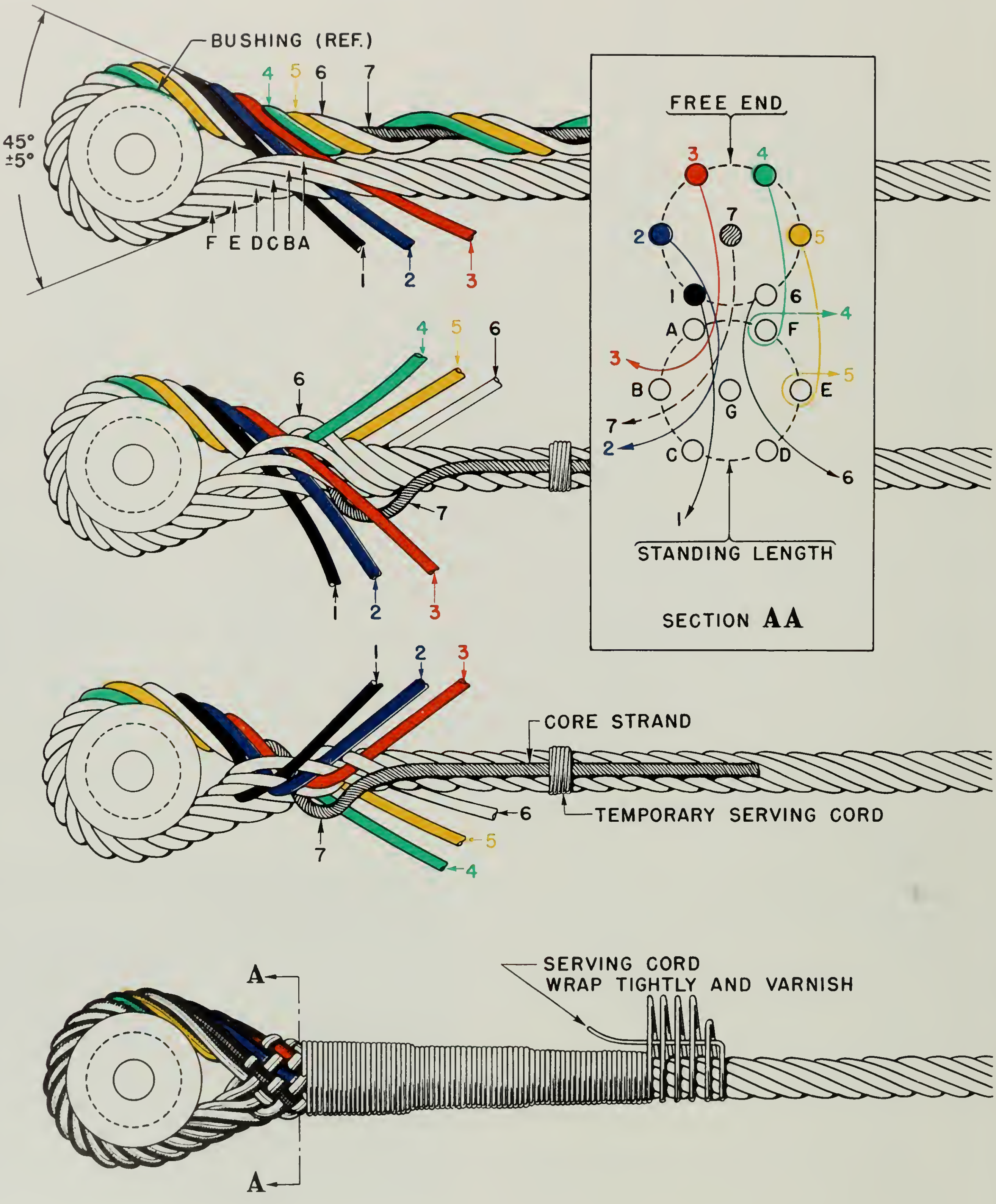


Figure 4—Preparation of a Woven Cable Splice

accurately measure the over-all cable length. If, after swaging, the terminals have more than the allowable 1/2-degree bend, secure them in a vise and straighten with as few applications of pressure as possible.

9. SWEAT-SOLDERED TERMINALS

Sweat-soldered terminals should be employed in fabrication only where they were originally utilized. See Figures 1 and 2 for specific types of tinned terminals utilized in original fabrication. Stearic acid or a suitable mixture of stearic acid and resin or resin dissolved in alcohol may be used as the soldering flux. Muriatic acid should not be used as a flux. The terminals should be prepared as follows: Apply the soldering flux to the end of the cable and insert the end of the cable into the barrel of the terminal, allowing the cable to extend through the barrel a small amount. Free the end strands of the cable and allow them to fray. Pull the cable back into the barrel until it is flush, then thoroughly sweat solder into the cable and barrel until solder appears at both ends of the barrel. Avoid overheating the solder. Sweat-soldered terminals can be easily distinguished from swaged type terminals by the air holes provided in the barrel of the terminal. These air holes allow the molten solder to permeate the strands of the cable with no air bubbles attendant. *(See Figure 5.)*

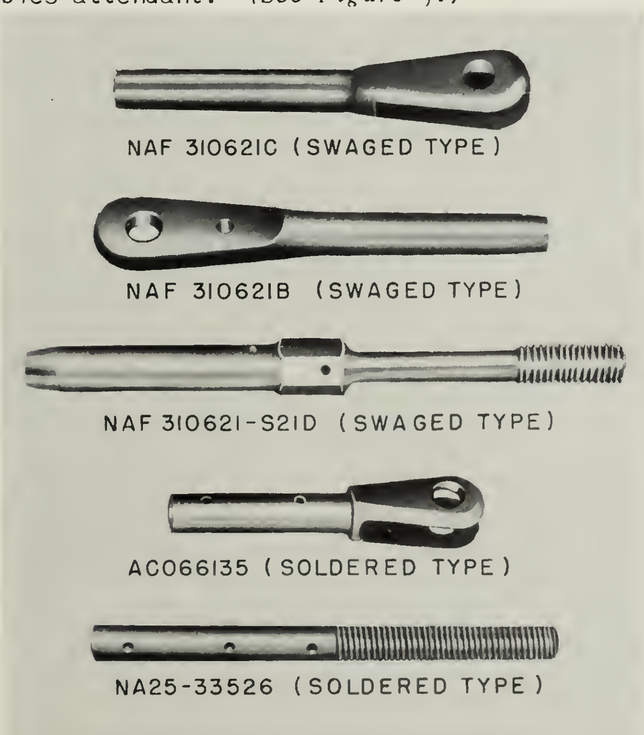


Figure 5—Types of Cable Terminals

10. WOVEN SPLICED TERMINALS

The five-tuck woven spliced terminals may be utilized on cables of 3/32-inch diameter or greater in place of swaged terminals where facilities are limited and immediate replacement is imperative. In some cases it will be necessary to splice one end of the cable on assembly. For this reason, investigate the original installation for pulleys and fairleads that might restrict the passage of the splice. The procedure for the fabrication of a woven splice is as follows: See Figure 4 for the designation of numbers and letters referred to in this sequence of operations, and see Paragraph 4 for fittings required.

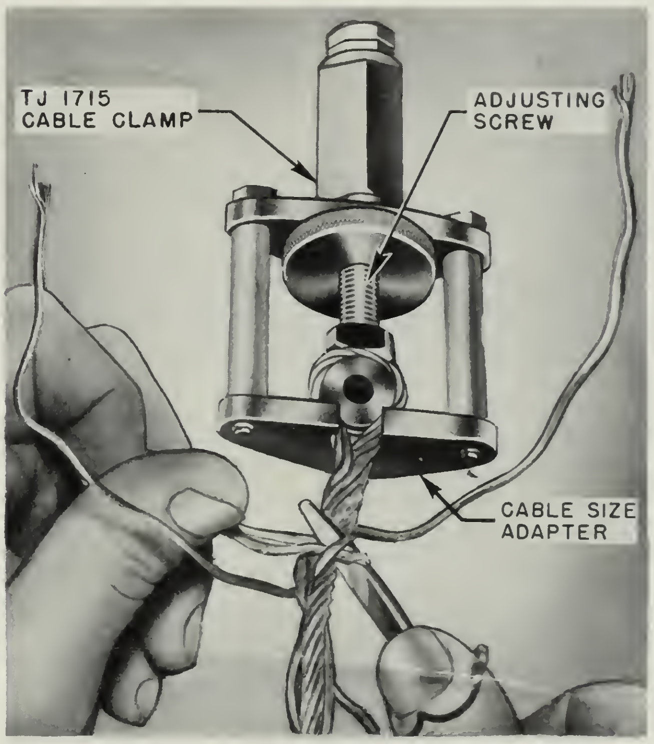


Figure 6—Cable Clamp for Woven Splice

1. Secure the cable around a bushing or thimble by means of a splicing clamp, leaving 8 inches or more of free end. Secure the splicing clamp in a vise with the free end to the left of the standing wire and away from the operator. *(See Figure 6.)* If a thimble is used as the end fitting, turn the points outward approximately 45 degrees.

2. Select the free strand (1) nearest the standing length at the end of the fitting and free this strand from the rest of the free ends. Next, insert a marlinspike under the first three

strands (A, B, C) of the standing length nearest the separated strand of the free end and separate them momentarily by twisting the marlinspike. Insert the free strand (1) under the three separated strands through the opening created by the marlinspike. Pull the free end taut by means of pliers.

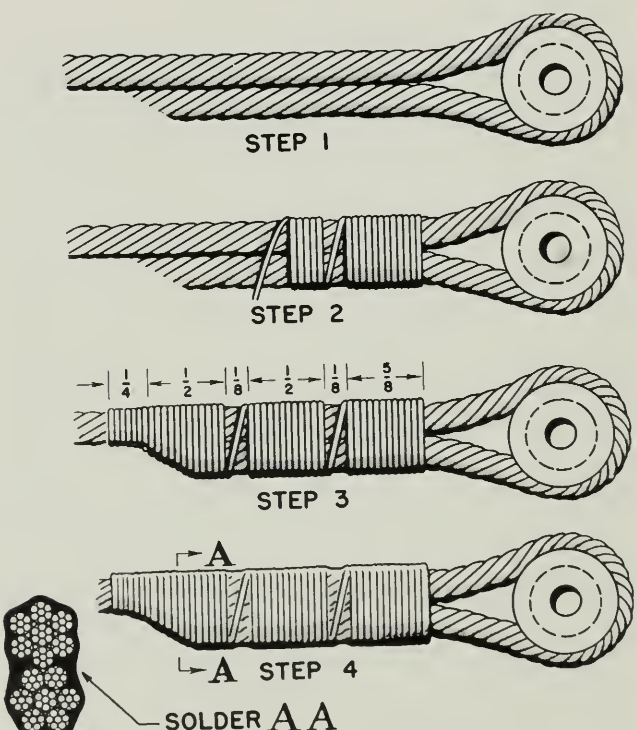
3. Unlay a second strand (2) located to the left of the first strand tucked, and insert this second strand under the first two standing strands (A,B). Loosen the third free length strand (3) located to the left of the first two, and insert it under the first standing strand (A) of the original three *(Detail A)*.

4. Remove the center or core strand (7) from the free end, and insert it under the same standing strands (A,B). Temporarily secure the core strand to the body of the standing cable *(Detail B)*. Loosen the last free strand (6) located just to the right of the first (1) and tuck in under the last two strands (E,F) of the standing cable. Tuck the fifth free end (5) around the fifth standing strand (E). Tuck the fourth free end (4) around the sixth standing strand (F) *(Details B,E)*. Pull all strands snug toward the end fitting with the pliers. This completes the first tuck.

5. Begin with the first free strand (1) and work in a counterclockwise direction, tucking free strands under every other standing strand. After the completion of every tuck, pull the strands tight with pliers. Pull toward the end fitting *(Detail C)*. After the completion of the third complete tuck, cut in half the number of wires in each free strand. Make another complete tuck with the wires remaining. At the completion of the fourth tuck, again halve the number of wires in the free strands and make one final tuck with the wires remaining. Cut off all protruding strands and pound the splice with a wooden or rawhide mallet to relieve the strains in the wires. Serve the splice with waxed linen cord. Start 1/4-inch from the end of the splice and carry the wrapping over the loose end of the cord and along the tapered splice to a point between the second and third tucks. Insert the end of the cord back through the last five wrappings and pull snug. Cut off the end; and if a thimble is used as an end fitting, bend down the points. Apply two coats of shellac to the cord, allowing two hours between coats *(Detail D)*. Carefully inspect the cable strands and splices for local failure. Weakness in a woven splice is made evident by a separation of the strands of serving cord.

11. WRAP-SOLDERED TERMINALS

On cables of 1/16-inch diameter only, the wrap-soldered splice may be employed to fabricate end fittings. See Figure 7 and Paragraph 4 for method and materials. Stearic acid or a suitable mixture of stearic acid and resin or resin dissolved in alcohol may be used as the soldering flux. Muriatic acid should not be used as a flux. Arrange the cable and the fittings as required, allowing approximately 2-1/4 inches of free end. Place the assembly in a splicing clamp and secure in a vise. Starting as close as practicable to the end fitting, press the free end standing lengths of the cable together tightly and wrap with a single layer of No. 20 brass or copper wire, leaving a space of approximately 1/8-inch between every 1/2-inch of wrapping. Care must be exercised to prevent the standing length from twisting during this operation. Allow the wrapping to extend approximately 1/4-inch beyond the free end. Dip the wrapping in tin-lead solder and carefully sweat the solder into the cable and about the wrapping. Apply the solder until the wrapping wire is barely discernible and make certain that the open spaces between the wrap sections are thoroughly impregnated with



Title 7—Wrap Soldered Splice

solder. After the splice has cooled, thoroughly wipe clean and positively remove all soldering flux by washing in hot water. Wipe the cable and impregnate the spliced section with Corol No. 95 or equivalent. Carefully inspect the splice. A wrap-soldered splice, easily bent with the fingers, is unsatisfactory because of a lack of solder penetration. Cracks in the solder located between the wrapping wire and the short space provided between wraps is a positive indication of slippage in the wrap-soldered splice.

12. TESTING

All cables and splices should be tested for proper strength prior to installation. Arrange cables to simulate installation, including pulleys where required, and gradually apply a load to one end of the cable for a period of 3 minutes. A suitable guard should be placed over the cable while it is being tested, to prevent personal injury in the event of cable failure. Apply test load to cables as follows:

Cable Size	Load in Pounds
1/16	300
3/32	550
1/8	1150
5/32	1550

13. COLOR BANDING

All cables fabricated locally should be color coded in accordance with the instruction given in Section

14. TOOLS

In connection with the preparation of fabricated cables, the following tools may be required: (1) a small pointed marlinspike of oval cross section, (2) a pair of side cutting pliers, (3) a splicing clamp *(see Figure 6)*, (4) a rawhide or hard wood mallet, (5) a die-swaging machine. *(See Figure 3.)*

SECTION 8

ELECTRICAL SYSTEM

1. GENERAL.

Refer to NA Dwgs. 77-54001, 84-54001, and 88-54001 for general installation of electrical equipment and the various main assembly component part numbers for the AT-6A, AT-6B, and AT-6C Airplanes, respectively. For radio equipment installation, refer to NA Dwg. 77-71001 for the AT-6A and AT-6B Airplanes and to NA Dwg. 88-71001 for the AT-6C Airplane. For wire routing, length, and size and for terminal types and sizes, refer to NA Dwgs. 77-54002, 84-54002, and 88-54002 for their respective models. Radio wiring charts are separate from the electrical wiring diagrams and will be found on NA Dwg. 77-71002 for the AT-6A and AT-6B, and on NA Dwg. 88-71002 for the AT-6C Airplane. Electrical and radio wiring diagrams for the particular model are furnished with each airplane and are stored in the data case in the cockpit. Concluding installation, ascertain that interiors of junction and switch boxes are free from filings, cuttings, bits of solder, pieces of tape, and pieces of wire, the presence of which is cause for shorting of circuits and the lowering of the dielectric strength of the insulation. Ascertain that terminals, terminal nuts, and conduit fittings and clamps are secure.

2. WIRING.

All wiring should be installed in a workmanlike manner. Wiring within conduit should be installed neatly and every effort made to prevent crossing of wires inside rigid or flexible conduit. Wiring must be of gage specified on the electrical wiring diagram and should be free of kinks, frays, and ruptures. Installation of wiring should be such that all tension on wires is eliminated. Wires within junction and switch boxes and on all items of electrical equipment should be routed to provide sufficient wire length to facilitate reterminaling in the event such becomes necessary. *(See Figure 1.)* Cord lacing and spot tying of wires and wire groups are required within all switch and junction boxes and on switch box covers. See Figure 6, Items 4 and 5, for typical examples. Where a number of wires enter a box, individual groups leading from each conduit should be laced separately to facilitate wire renewal. The most satisfactory method of lacing a wire group is accomplished by the use of the marlin, or hammock, hitch. This hitch is the reverse of a common half hitch. The cord passes over-and-under the loop instead of under-and-over as it does in a half hitch. *(See Figure 6, Item 5.)*

3. WIRE NUMBERING.

When reterminaling or replacing a wire, care must be taken to renumber the wire properly. If no identification number exists on the wire before repair, or if the number is illegible,

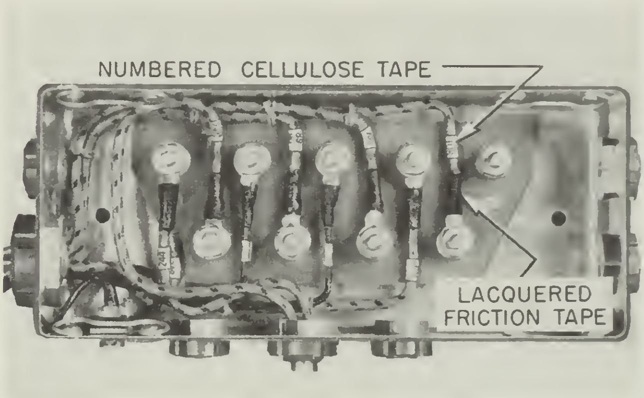


Figure 1—Typical Wire Routing

ascertain the correct number by carefully checking the electrical wiring diagram. Original identification numbers are cut from Minnesota Mining Company numbered cellulose tape. *(See Figure 1.)* However, if this tape is not available, a facsimile may be made by the use of white paper and clear cellulose tape. Wrap a minimum of 1-1/2 turns of tape around the wire and apply one coat of clear lacquer over the tape.

4. WIRE REPLACEMENT.

Damaged wires should be replaced with new wires. Before removing the damaged wires or wire group, tie a strong waxed cord to one or more of the wires to be removed. By pulling the wires out the opposite end, the cord is threaded through the conduit. After the damaged wires are pulled out of the conduit, the cord may be utilized to pull the new wires

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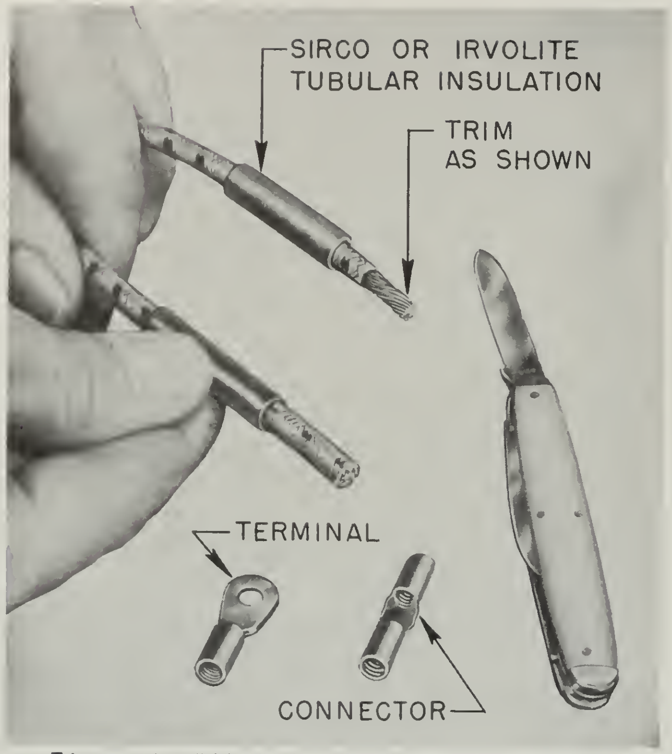


Figure 2—Slipping Insulation Over Wire

Figure 3—Placing Terminal Over Bared Wire

back through the conduit and into position. If a conduit repair is to be made, this method cannot be used. However, after the conduit repair is completed, a stiff resilient wire, such as piano wire, may be utilized to pull the wires or waxed cord through the conduit. If this method fails or if wire is not available, a string may be blown through the conduit. Knot the string and place the knot a few inches within the conduit. Place a compressed air nozzle at the opening and feed the string into the conduit while allowing the compressed air to blow the string through the conduit and out the opposite end.

5. WIRE TERMINALS.

Two types of terminals are employed on the electrical wiring, the soldered and the crimped type. The terminals used in repair work must be of the size and type specified on the electrical wiring diagram. *(See Figure 7.)* The types of terminals illustrated are used on the AT-6A, AT-6B, and AT-6C Airplanes. When removing insulation prior to terminaling, care must be exercised not to nick individual strands of the conductor. In case a soldered terminal is to be used, avoid overheating the wire and applying excessive solder. Use a tin-lead solder and a resin base flux. Acid base flux lowers the dielectric strength of the insulation materials. Shoulders of the terminals should be taped and coated with clear lacquer to prevent short-circuiting by contact with adjacent terminals.

6. ATTACHING CRIMPED TERMINALS.

To attach crimped-type terminals or connectors to the electrical wiring, proceed as illustrated in this Section. *(See Figures 2, 3, 4, and 5.)* Trim the insulation from the wire for a length of 1/16-inch plus the length of the terminal shoulder. Slip the appropriate length of Sirco or Irvolite tubular insulation over the wire before attaching the terminal. Slip the terminal or connector shoulder over the bared wire end as shown. *(See Figure 3.)* Make certain that all the wire strands are inside the tubular shoulder. Using the special Sta-Kon crimping pliers, crimp the shoulder of the terminal or connector and slip the Sirco or Irvolite insulation down over the shoulder of the terminal as shown. *(See Figures 4 and 5.)* If neither of these insulators is available, tape the terminal with friction tape and apply a coat of lacquer. Sta-Kon terminal crimping pliers are available in three sizes: size No. WT-112 for use on Types A and B terminals; size No. WT-113 for use on Type C terminals; size No. WT-114 for use on Types D, E, and F terminals.

7. CONDUIT.

For dimensions, part numbers, and routing of all rigid and flexible conduit, refer to NA Dwgs. 77-54001 and 77-71001 for the AT-6A Airplane, NA Dwgs. 84-54001 and 77-71001 for the AT-6B Airplane, and NA Dwgs. 88-54001 and 88-71001 for the AT-6C Airplane. Sizes specified on these drawings must be adhered to. Conduit should be supported by means of suitable clamps or clips, as specified on the respective drawing for that model airplane. Under no circumstances should cord lacing be utilized as a means of conduit support or attachment. Conduit should be free from dents, scratches, and/or other blemishes. Care should be exercised when installing conduit to ascertain that conduit enters box fitting or bulkhead union properly and not at an angle. Do not attempt straightening of conduit by tightening fitting nuts, as this action will dent or bend the conduit if the conduit is of a small diameter. If the conduit is thick-walled and of a large diameter, the side of the junction box will be pulled out of shape. Concluding installation, ascertain that conduit fittings are secure and that all conduit has clearance at all frame cutouts and at all other sharp or protruding members. Conduit must be bonded by means of removing the finish at all unions, box fittings, and under all supporting clamps. Methods of splicing conduit which has been damaged are outlined under tubing repairs in Section IX.

8. BONDING.

Electrically isolated metal parts and shielding should be bonded by interconnecting these parts to the airplane structure. Connections for bonding must be of good electrical value, offering resistance not to exceed .001 ohm at each connection. Bonding should be accomplished by soldering, bolting, clamping, or welding as required. Metal-to-metal joints permanently joined by welding, soldering, sweating, or riveting require no further bonding. No bonding is required for small isolated parts such as screws and small coverplates, or for control cables, insulated lines or members less than 60 inches long, or permanently isolated members subject to unlimited travel. With the exception of those located forward of the firewall, flexible and rigid conduit of the AT-6C is bonded by the clamps which support it, and on the AT-6A

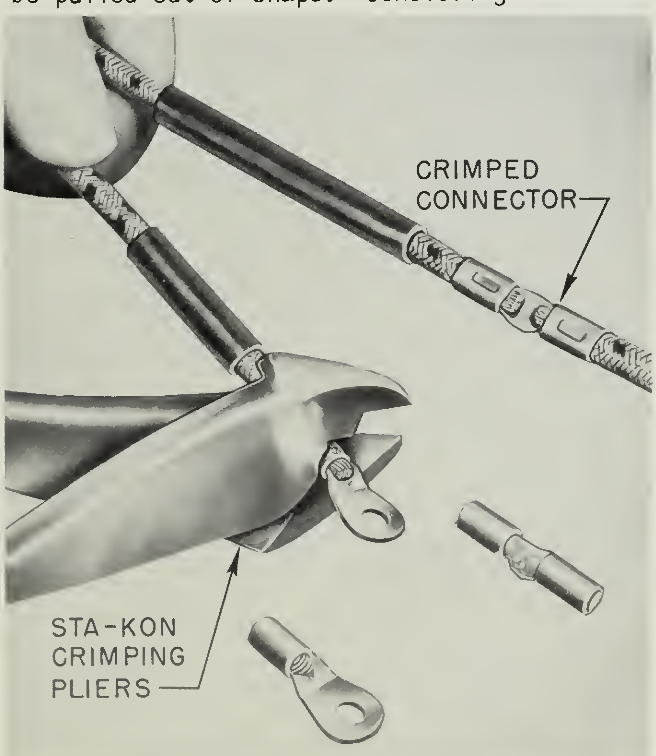
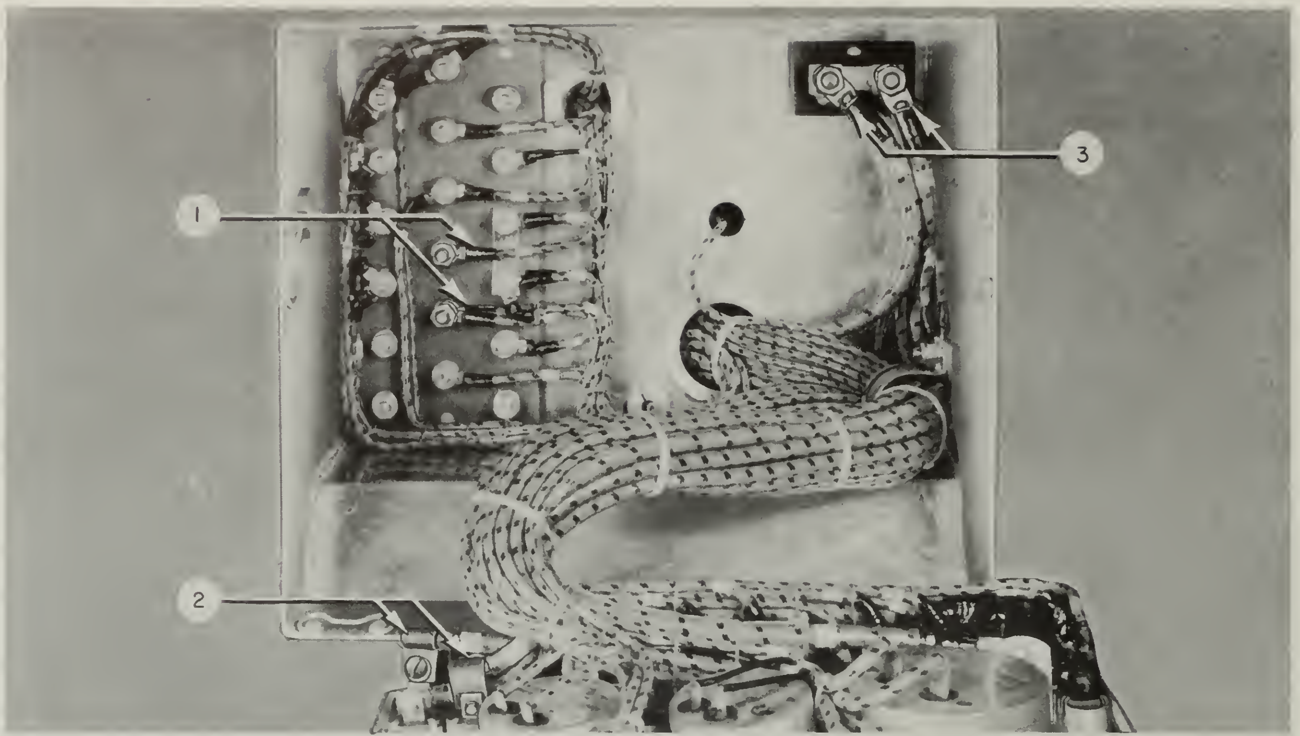
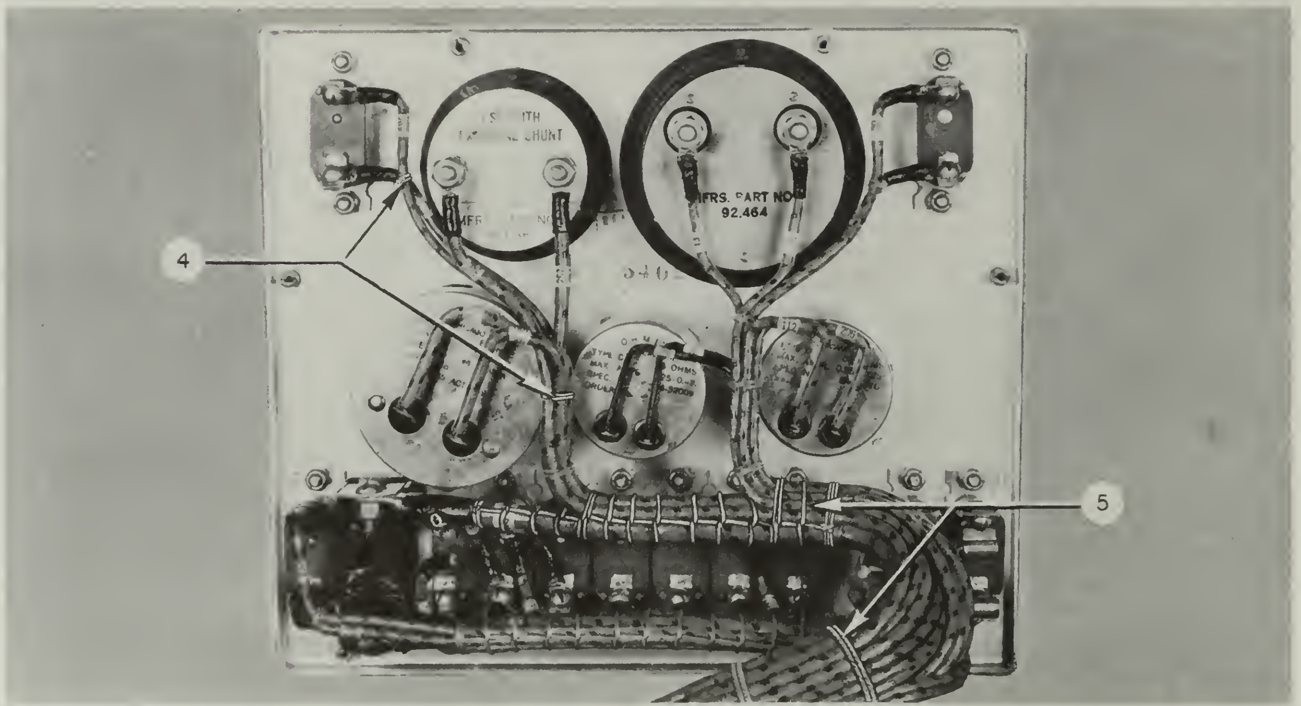


Figure 4—Crimping the Terminal

Figure 5—Slipping Insulation Into Position



SWITCH BOX INTERIOR



SWITCH BOX COVER

1. AN 660 TERMINALS — LIGHT TYPE
2. AN 660 TERMINALS — ROLLED TYPE
3. AN 660 TERMINALS — HEAVY TYPE
4. SPOT TIES
5. WIRE GROUP LACING

Figure 6—Typical Switch Box Wiring

and AT-6B by tinned copper flexible bonding braids, Type B1209. *(See Figure 8.)* In ordering these bonds, the first dash number is the terminal size on one end; the second dash number is the terminal size on the opposite end; the third dash number is the width of the bond in 1/8-inch; and the fourth and fifth dash numbers designate the length of the braid measured from terminal hole to terminal hole. Example: A B1209-4-8-1-3-5 bonding braid has a terminal on one end for a No. 4 screw and a terminal on the other end for a No. 8 screw. The bonding braid is 1/8-inch wide and

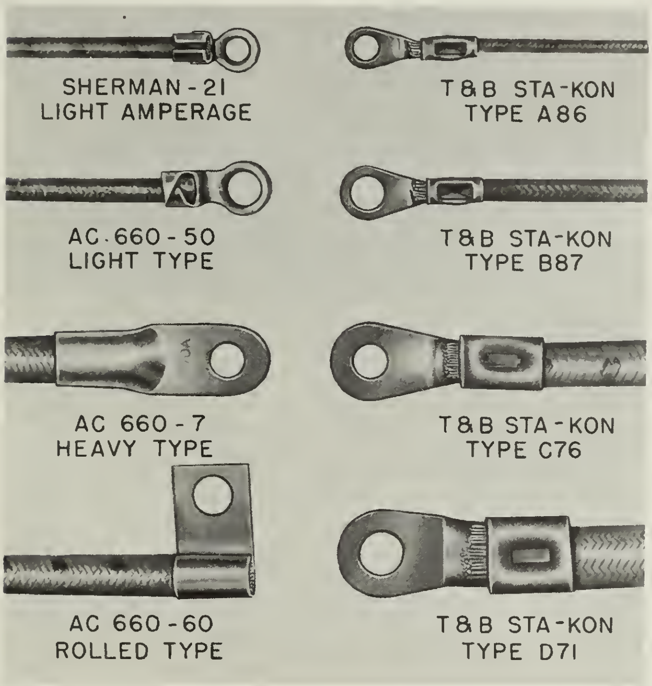


Figure 7—Types of Terminals

3-1/2 inches long. Attach the bonding braid to copper flexible conduit by means of sweat soldering and to rigid conduit by means of the welded bonding lugs provided. Attach the bonding braid to the rigid conduit only as illustrated with the terminal under the nut to serve as a shakeproof lock washer. *(See Figure 8.)*

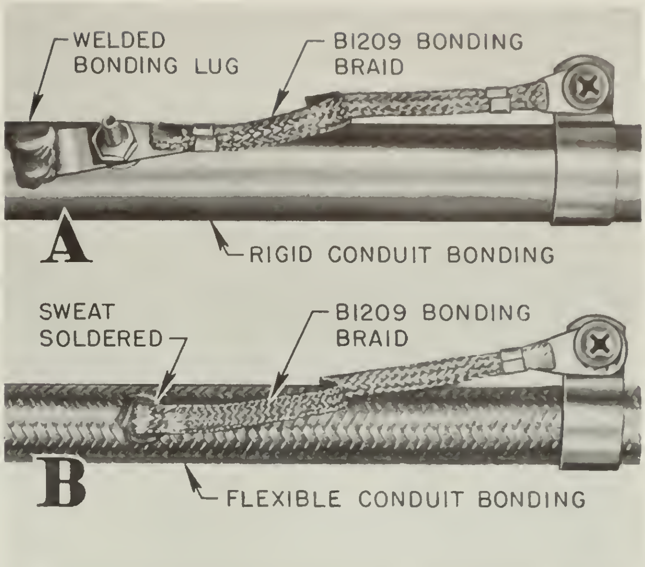


Figure 8—Conduit Bonding Methods

9. ELECTRICAL REPAIR TOOLS AND MATERIALS.

For electrical repairs, all or part of the following tools and materials are required:

TOOL OR MATERIAL

IRON, SOLDERING, 100 OR 200 WATT
PLIERS, CRIMPING, STA-KON, SIZE TO SUIT
TERMINALS, SOLDERED-TYPE, AC660, SIZE TO SUIT
TERMINALS, STA-KON, MADE BY THOMAS & BETTS CO., ELIZABETH, NEW JERSEY, SIZE TO SUIT
TAPE, FRICTION, 1/2-INCH WIDE, PROCURE LOCALLY
CORD, LACING, STRONG WAXED, PROCURE LOCALLY
WIRE, SOFT SOLDER, 50/50 TIN-LEAD
FLUX, SOLDERING, RESIN BASE, PROCURE LOCALLY
TUBING, SIRCO INSULATION, MFG. BY SUPRENANT ELECTRIC INSULATION CO., BOSTON, MASS.
TUBING, IRVOLITE INSULATION, MFG. BY IRVINGTON VARNISH & INSULATOR CO., IRVINGTON, N.J.
LACQUER, CLEAR, PROCURE LOCALLY
TAPE, CELLULOSE, CLEAR OR NUMBERED, MFG. BY MINNESOTA MINING & MFG. CO., ST. PAUL, MINN.
BRAID, BONDING, B1209, LENGTH & TERMINALS TO SUIT

1. WIRE STRIPPERS
2. DIAGONAL CUTTERS
3. LONG NOSE PLIERS
4. CRIMPING PLIERS
5. FORMING PLIERS
6. WIRE HOLDING CLAMP FOR PULLING ON INSULATION
7. MIDGET SOCKET WRENCH SET
8. PHILLIPS HEAD SCREWDRIVER
9. STANDARD SCREWDRIVER
10. SOCKET EXTENSION HANDLE
11. SOLDERING IRON

Figure 9—Electrical Repair Tools

1. SOLDERING FLUX
2. SPOT-TYING CORD
3. LACING CORD
4. CLEAR LACQUER
5. WIRE SOLDER
6. LINEN SERVING CORD

7 & 8. CRIMPED TYPE TERMINALS

9, 10 & 11. SOLDERED TYPE TERMINALS

12. NUMBERED CELLULOSE TAPE
13. FRICTION TAPE

Figure 10—Electrical Repair Materials

SECTION 9

MISCELLANEOUS

TUBES AND TUBING REPAIRS

1. GENERAL.

Practically all tubing in this airplane is made of 52SO and 53SW aluminum alloy. A relatively small number of lines are made of copper, corrosion-resistant steel, and 2S-1/2H aluminum alloy. The tubular lines also vary in size, shape, and end finish. Damaged lines should be replaced with standard parts from North American Aviation, Inc., Inglewood, California, U.S.A. However, replacement is sometimes impractical, and therefore a repair method is outlined. Repair procedures described in the following paragraphs are applicable to all lines, regardless of their metal composition. Small scratches and abrasions, or minor forms of corrosion occurring on the exterior, after having been smoothed out with a burnishing tool or steel wool, may be considered as negligible.

2. TUBE JOINTS.

Two types of joints are used at pressure tube connections. The beaded or upset type of joint is used in the vacuum, fuel, and oil systems where hose connections are utilized. All other joints are of the flared type, to be used in conjunction with tube connector fittings. Grip dies and flaring or beading tools are required to form flared and beaded-type joints.

3. FORMING FLARED TUBE JOINTS.

The TJ 4411 grip die and flaring tool *(see Figure 4, Detail B)* is recommended for forming flared tube joints. The flared-type grip die consists of two steel blocks placed side by side and held in alignment by three steel pilot pins pressed into one block and extending into corresponding holes in the other block. *(See Figure 1.)* A number of countersunk holes are drilled along its length, the holes varying in size to correspond to the tube sizes. The grip die should be designed to grip the tubes tightly without damaging them. The flaring tools for this type of joint consist of cylindrical bars tapered at one end to correspond to the angles of the countersunk holes in the die. To make a flared tube end, insert the end of the tube into the grip die through the space provided when the two blocks are parted. Allow tube to extend upward through the die approximately one-half the outside diameter of the tube, and secure the assembly in a vise. Rest the tapered end of the flaring tool in the tube and tap lightly with a hammer until the walls of the tube are forced to assume the shape of the countersunk hole in the die. *(See Figure 1.)*

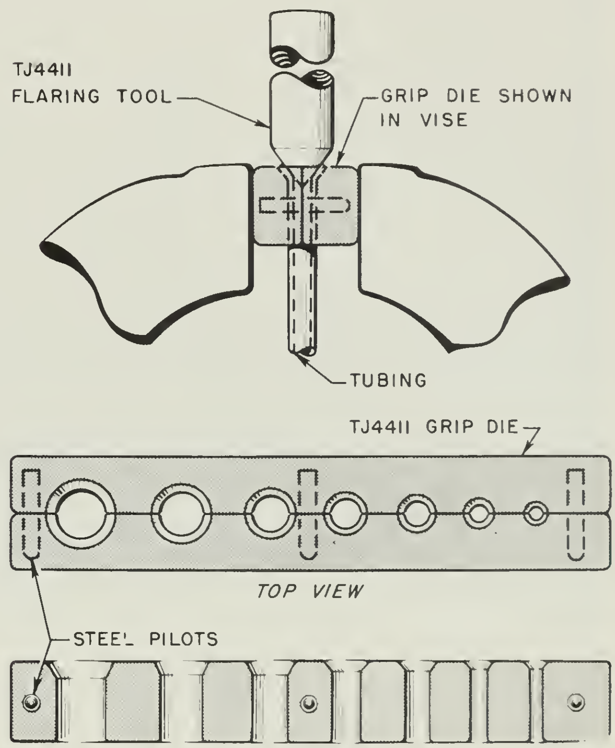


Figure 1—Tube Flaring Tools

4. FORMING BEADED TUBE JOINTS.

When making a beaded-type joint, a separate grip die should be used for each size tube. This grip die consists of two steel blocks placed side by side and held in alignment by suitable pilot pins. A hole, equal to the outside diameter of the pipe and chamfered slightly around the top, is drilled down the center of the die. *(See Figure 2.)* The beading tool consists of a cylindrical rod with a chamfered hole equal to the outside diameter of the tube drilled approximately 1/4-inch into one end of the

9

center. A hole, equal to the inside diameter of the tube, is drilled down through the center of this hole and the rod to a depth of 1 inch. A steel pilot is driven into this hole and permitted to extend approximately 1 inch below the tool. To make a beaded joint, insert the tube into the die through the space formed when the two blocks are parted, allowing the pipe to extend upward from the die approximately 1-1/2 diameters. Secure the assembly in a vise. Place the pilot of the beading tool into the open end of the tube and strike lightly until the desired bead is formed. *(See Figure 2.)* An alternate method of beading tubes of large diameter, utilizing a hand roller, is illustrated in Figure 3.

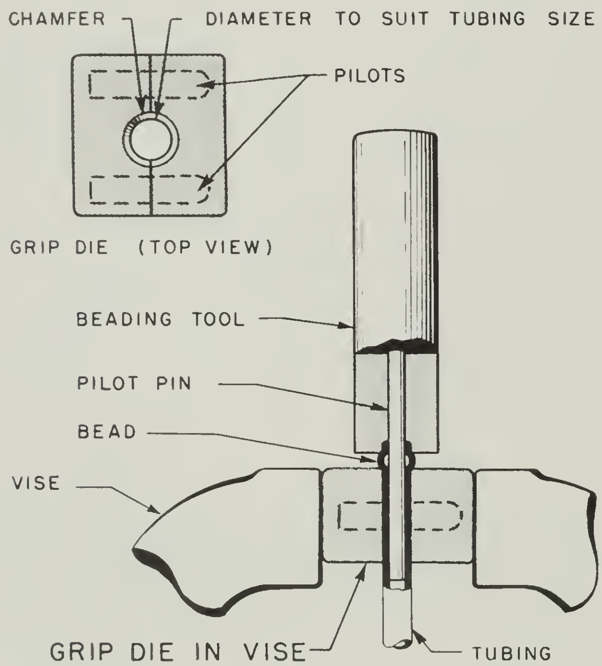


Figure 2—Beading Small Tubing

5. GENERAL TUBING REPAIR.

For the purposes of repair, the tubes may be divided into three groups consisting of: the high-pressure, flared end group; the low-pressure, beaded end group; and the conduit group which retains no pressure and requires no end finish. All tubing is repaired by replacing the damaged section with a new length of similar material. The length of tubing removed will be controlled by the location of the damage, the extent of the damage, and the most convenient location for tool manipulation. When the damaged portion of the line is near a fitting, union, or connection, cut the line beyond the point of damage and remove the length of line which contains the injury. If damage is not near a fitting, the line must be cut on both sides of the damage. Lines must be cut in locations where cutting, burring, and other operations can be accomplished. The use of a keyhole hacksaw is recommended for cutting the tubing.

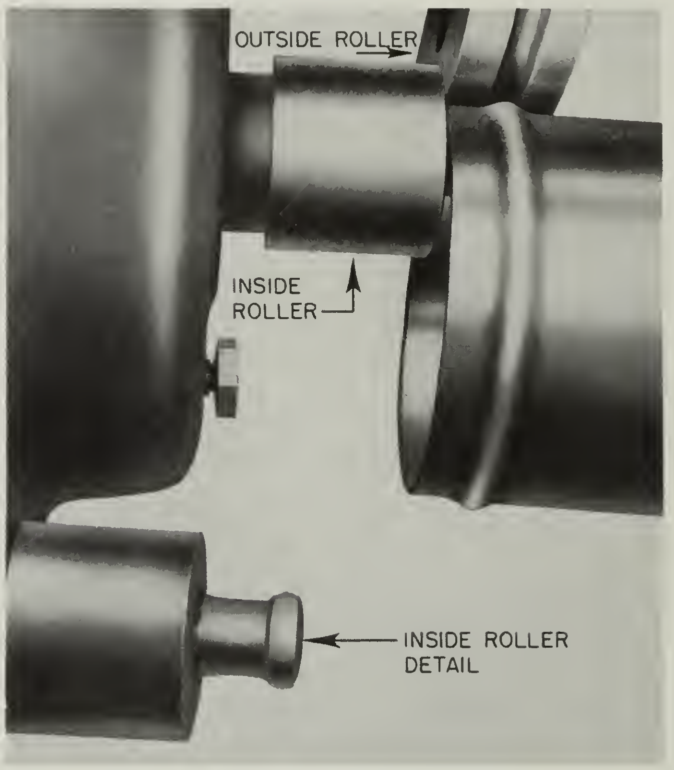


Figure 3—Beading Large Tubing

6. REPAIR OF HIGH-PRESSURE TUBING.

Cut out and remove damaged portion of tube as directed in the preceding paragraph *(see Figure 4, Detail A)*. Burr the ends of undamaged tube portions, and clean. Slip an AC- 11-BT nut and an AC-811-T sleeve of proper size and material onto the remaining tube ends. Flare the tube ends using a TJ 4411 grip die and a flaring tool of proper size and a hammer *(see Figure 4, Detail B)*. Support rear side of grip die to prevent tube bending when hammer thrusts are applied. Tap the flaring tool into end of tube until the walls of the tube assume the shape of the countersunk hole in the die. Insert an AC-811-HT union of proper diameter and material into flared tube ends. Apply Sealube to threads and slip sleeve down against flared tube end. Thread the nut onto the union and tighten. *(See Figure 4, Detail C.)* Measure accurately the distance between the union faces *(length "E", Detail C, Figure 4)*. If the length of the damage is several inches or the damaged tubing is to be removed to the nearest fitting, a repair insert must be used. Cut a repair length of similar tubing; clean and burr the ends; slip on two sleeves and two nuts in proper sequence; and flare the tube ends as directed in Paragraph 3. Completed repair insert is shown in Figure 4, Detail D. Place

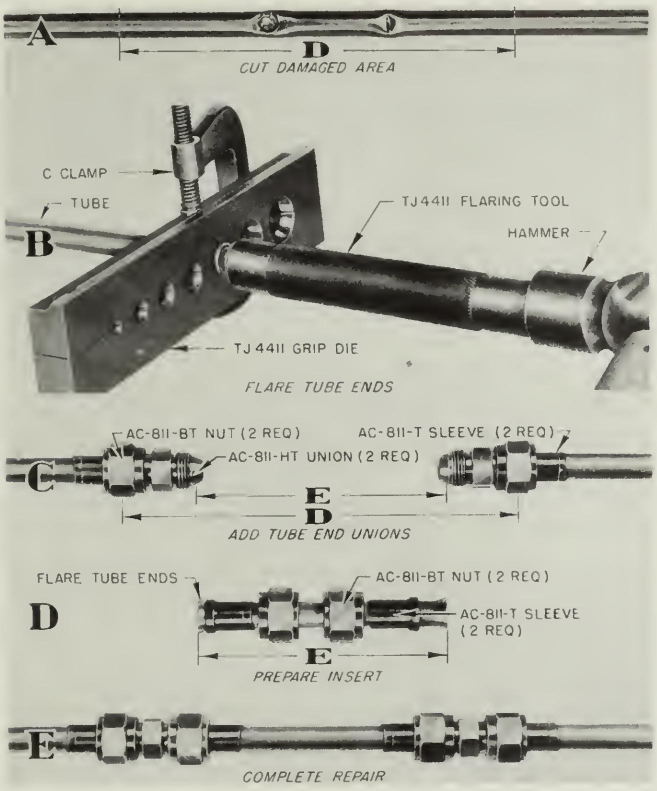


Figure 4—High-pressure Tubing Repair of Major Damage

this repair section between the two unions; thread on the two nuts and tighten. *(See Figure 4, Detail E.)* The use of special wrenches adapted for the hex nuts on high-pressure line connections will, in many cases, expedite repairs. Occasions arise when restrictions prohibit the use of open end wrenches. *(See Figure 8, Detail A, for two types of wrenches and Detail B for their use.)* If the damaged portion of the line does not exceed the length of a union, a repair section is not needed. Cut out the damage as directed in Figure 5, Detail A, so that after the tube ends are flared as shown on Figure 4, Detail B, and a union is inserted, a tight fit will result. *(See Figure 5, Detail D.)*

7. LOW-PRESSURE LINE REPAIR.

Remove the damaged portion of the tube as directed in a foregoing paragraph, burr the remaining tube ends, and remove cuttings from tube interiors. *(See Figure 6, Details A and B.)* If possible, bead the remaining tube ends as outlined in Paragraph 4. Cut a repair section 1/2-inch shorter than the length of the removed damaged portion. Burr the section and bead as prescribed in a foregoing paragraph, and

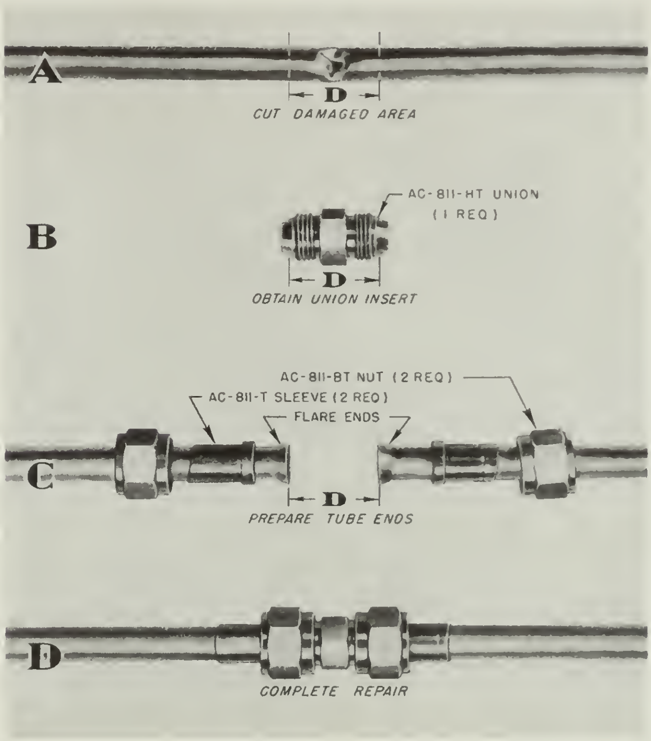


Figure 5—High-pressure Tubing Repair of Minor Damage

clean. *(See Figure 6, Detail C.)* Cut hose connections of proper length and diameter from synthetic rubber hose and slip two hose clamps over the ends of each. Slip one hose connection well back over original tube before placing repair insert into position. Slip hose connections midway over the junction formed by the original tube and the repair section as illustrated in Figure 6, Details D and E. Tighten clamps. Glycerin applied to a metal pipe aids the sliding of rubber hose.

Note: Two hose clamps on each end of a connection are normal on some beaded oil lines, as these lines maintain a relatively high pressure.

8. CONDUIT REPAIR.

Before starting repair to conduit, remove all wires from the tube interior. Cut out the damaged portion of the conduit in accordance with Paragraph 5, and as illustrated in Figure 7, Detail A. Burr and clean the remaining tube ends. *(See Figure 7, Detail B.)* Cut an insert of identical tubing the same length as the removed damaged portion. Burr the ends and clean the tube interior. *(See Figure 7, Detail C.)* Place a conduit union on each end of the repair insert and place this assembly between the remaining ends of the original tube. If rigidity of the original tube ends prohibits this type of insertion, one or both union sleeves may be cut in half, opposite normal slit. *(See Figure 7, Detail C.)* Place conduit nuts over the tubing before placing repair insert into position. Apply a thin coating of threadlube to the union threads. Hold union halves in position around the junction formed by the insert and the original tube end; screw on conduit nuts and tighten. Completed repair is shown in Figure 7, Detail D.

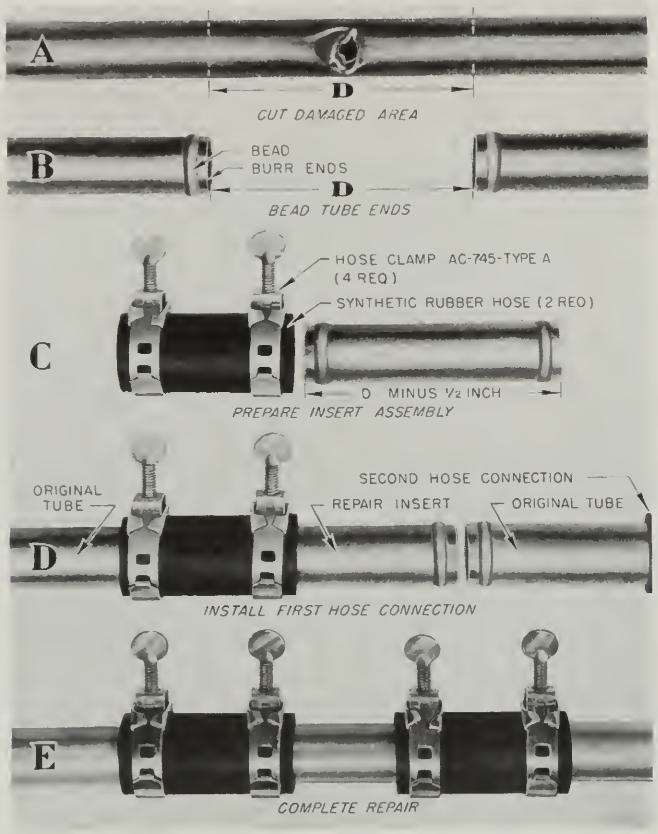


Figure 6—Low-pressure Tubing Repair

9. MISCELLANEOUS.

Various lines such as instrument, drain, and overflow lines fall under the high-pressure and the low-pressure groups. Although some of these maintain little or no pressure, they must be treated and repaired as high-pressure lines, as no leaks are permitted. If the fittings of the damaged line are of the flared type, repair as outlined in Paragraph 6. If the lines are assembled by hose connections, repair as outlined in Paragraph 7.

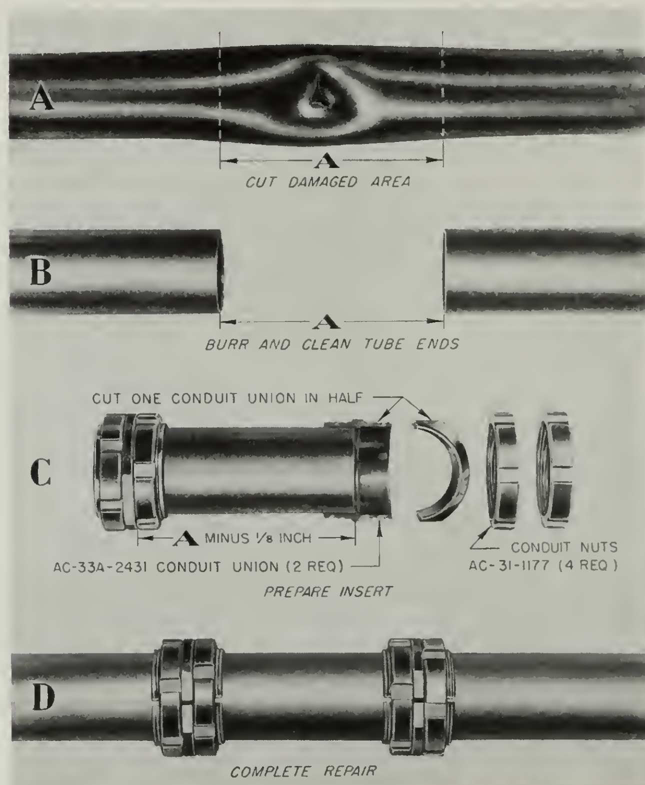


Figure 7—Conduit Repair

10. TOOLS.

All or part of the following tools are necessary to complete tubing repairs. All tools having a TJ part number are made by, and can be purchased from, North American Aviation, Inc., Inglewood, California, U.S.A.

TOOLS	REMARKS
WRENCH, SPECIAL "B" NUT TJ 2802	LOOSENING OR TIGHTENING AC-811-BT NUTS. ALL REQUIRED SIZES AVAILABLE. *(See Figure 8.)*

TOOLS	REMARKS
WRENCH, OPEN-END TJ 601	LOOSENING OR TIGHTENING HIGH-PRESSURE AND/OR CONDUIT FITTINGS. SIZES RANGE FROM 1/2-INCH TO 1-1/2 INCHES.
GRIP DIE OR FLARING TOOL TJ 4411	USED TO FLARE TUBE ENDS.
HACK SAW, SMALL OR KEYHOLE	CONVENIENT FOR CUTTING TUBING IN RESTRICTED PLACES.
WRENCH, CONDUIT NUT TJ 2488	LOOSENING OR TIGHTENING CONDUIT NUTS. DIAMETERS OF 1/2-INCH TO 1-1/2 INCHES AVAILABLE.
TOOL, BEADING, (SEE FIGURE 2.)	HAND BEADING TUBES OF DIAMETERS UP TO 5/8-INCH.
WRENCH, CONDUIT NUT TJ 2332	USED TO GRIP CONDUIT NUTS.
BEADING TOOL, HAND ROLLER (SEE FIGURE 3.)	FOR BEADING TUBING FROM 5/8-INCH UP.

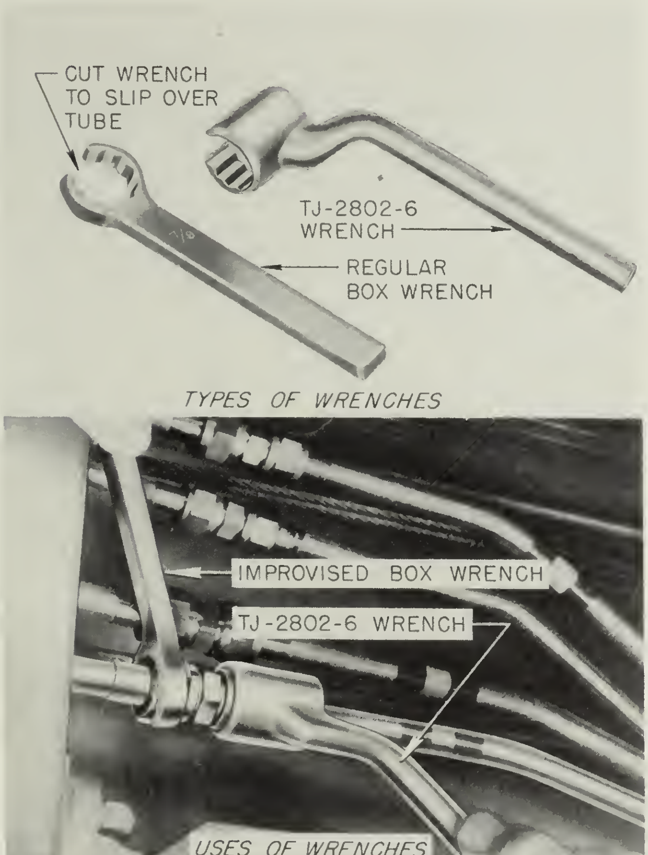


Figure 8--Special Wrenches for High-pressure Connections

11. MATERIALS.

The following is a list of materials which may be required to accomplish the three types of tubing repair.

MATERIALS	REMARKS
TUBING, REPAIR LENGTHS	MUST BE IDENTICAL TO DAMAGED TUBING IN MATERIAL, DIAMETER, AND WALL THICKNESS.
TUBING, SYNTHETIC RUBBER	MUST BE IDENTICAL TO OTHER CONNECTIONS FOUND ON THE LINES OF THE SYSTEM BEING REPAIRED.
NUT, AC811-BT	
SLEEVE, AC811-T	HIGH-PRESSURE, FLARED-TYPE FITTING REPAIR.
UNION, AC811-HT	
CLAMP, HOSE	AN745, TYPE A MINIMUM DIAMETER AVAILABLE 9-1/6 INCHES. MAXIMUM DIAMETER AVAILABLE 3-1/4 INCHES.
CLAMP, HOSE	AN745, TYPE B MINIMUM DIAMETER 1/2-INCH. MAXIMUM DIAMETER 1-1/8 INCHES.
NUT, CONDUIT	AC31-1177 USED ON CONDUIT UNIONS.
UNION, CONDUIT	AC33A-2431 AVAILABLE IN SIZES FROM 1/4-INCH DIAMETER TO 2 INCHES.

CHERRY BLIND RIVETS

12. GENERAL.

Cherry blind rivets, manufactured by the Cherry Rivet Company, Los Angeles, California, offer a practical solution to the problem of blind spot riveting, as they require no bucking.

Designed to take care of the difficult riveting jobs in inaccessible places or in double-surfaced structures where access to both sides of the work is impossible, this type rivet is recommended for repair work whenever standard riveting is impractical. Cherry rivets are available in two types, the hollow and the self-plugging type. Being the stronger of the two, the self-plugging type is the only one recommended for repair work. The shear strength of self-plugging Cherry rivets of 5/32-inch diameter and less is 60 percent of the standard rivet shear strength specified in the ANC-5 Manual (Strength of Aircraft Elements). For rivets of 3/16-inch and larger, the shear strength is 50 percent of that of standard rivets of the same diameter and material.

13. SELF-PLUGGING CHERRY RIVETS.

The self-plugging type has a stem or mandrel with an expanded section. This expanded section is pulled into the hollow member and stays in place when the rivet is applied. When this rivet, with the stem projecting from the head end, is placed in riveting position, the gun fits over the stem and exerts pressure on the head and a pull on the stem. Thus the preformed head on the stem is forced into the tail of the rivet. This expands the tail and tightens it down against the back of the pieces to be fastened together. The pull on the stem is continuous until the stem breaks and falls out on either side of the rivet. The remaining end of the stem may then be nipped off if a flush surface is desired. *(See Figures 12, 13, 14, and 16.)* The expansion of the self-plugging type is particularly effective in filling the hole and gives a good tight job with satisfactory shear and bearing values. Three diameters are available: 1/8-, 5/32-, and 3/16-inch. Each diameter is made in a series of grip lengths and, with the sizes regularly made, it is possible to handle total thicknesses of material from .030 to .265 inch.

14. TYPES OF SELF-PLUGGING CHERRY RIVETS.

Four types of self-plugging Cherry rivets are available which are recommended for repairs. These types are listed and described in the following table:

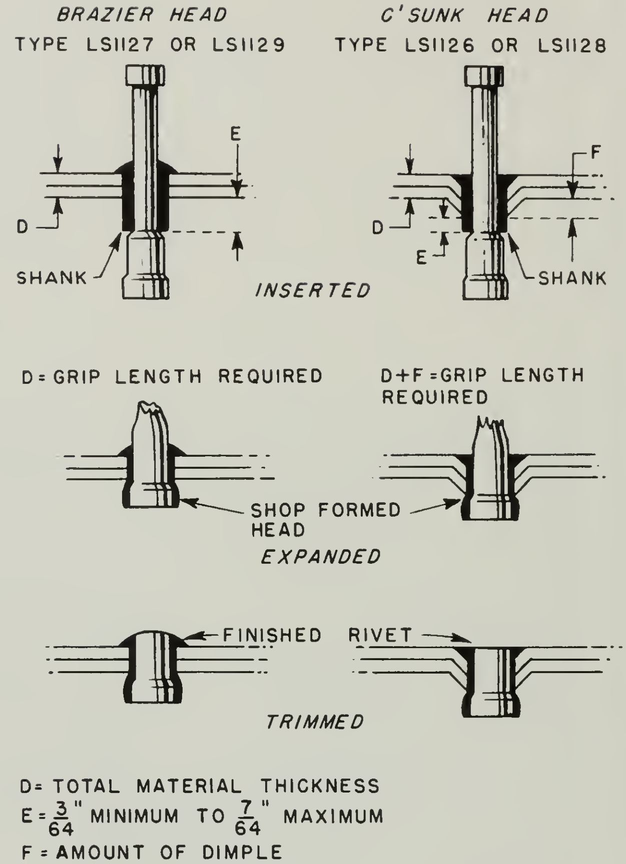


Figure 9—Cherry Rivet Grip Length

TYPE	REMARKS
LS1126 SERIES	THIS IS AN AD RIVET MADE OF A17ST ALUMINUM ALLOY AND REQUIRES NO ADDITIONAL HEAT TREATMENT. THE HEAD AND SHANK DIMENSIONS ARE IDENTICAL TO THE NEW 100-DEGREE COUNTERSUNK AN426 RIVET. THE PLUGGING STEM IS MADE OF 17ST ALUMINUM ALLOY, ADDING TO THE SHEAR VALUE OF THE RIVET.
LS1127 SERIES	THIS RIVET IS THE SAME AS THE LS1126 EXCEPT THAT THE HEAD AND SHANK HAVE THE DIMENSIONS OF THE WESTERN STANDARD BRAZIER.
LS1128 SERIES	THIS SERIES IS DESIGNED PRIMARILY FOR USE IN DOUBLE-DIMPLED CONSTRUCTION WHERE THE PILOT HOLE SIZE IN THIN SHEETS HAS BEEN ENLARGED FROM .010 INCH TO .020 INCH OVER THE NOMINALLY SIZED FRACTIONAL DIAMETERS. THE SHANK DIAMETER HAS BEEN ENLARGED

TYPE	REMARKS
	.099 INCH TO .016 INCH OVER THE NOMINAL FRACTIONAL DIMENSION IN ORDER TO FIT THE ENLARGED PILOT HOLES. TESTS TO DATE INDICATE THAT THIS RIVET WILL BRING THE SHEAR AND FATIGUE VALUES OF THE DIMPLED JOINT NEARLY EQUAL TO THE EQUIVALENT JOINT WITH SOLID BRAZIER HEAD RIVETS.
LS1129 SERIES	THIS SERIES HAS THE SAME SHANK DIAMETER AS LS1128, BUT HAS WESTERN STANDARD BRAZIER HEADS.

15. DETERMINING CHERRY RIVET GRIP LENGTH.

The grip length of Cherry rivets is measured in thirty-seconds of an inch and appears as the last dash number following the Cherry rivet type number. Example: LS1127-5-6 refers to a rivet with a grip length to handle material having a total thickness of 3/16-inch. The proper grip length to use is determined by measuring the total thickness of the sheets to be fastened together. Select a rivet which has a nominal grip length as close as possible to this measurement. There is, however, a considerable amount of tolerance, in that Cherry rivets will handle material which is 1/64-inch (.016 inch) thicker or 3/64-inch (.047 inch) thinner than the nominal or recommended grip length. The total length of an unexpanded rivet shank includes approximately 1/16-inch additional for proper heading. Therefore, to ascertain the grip length of a Cherry rivet in the event this information is unknown, measure the shank and subtract 1/16-inch. Where dimpled sheets are to be used in connection with countersunk rivets, determine the proper grip length by taking the total thickness of the sheets plus the amount which the dimple extends beyond the inside surface, or add the height of the head to the thickness of the sheets before dimpling. Example: If the sheets are .052 inch in total thickness and the dimple extends .038 inch on the inside of the work, then the total grip length to be handled would be .090 inch. This is .035 inch less than 1/8-inch which is within the limits covered by the 1/8-inch grip length. The required grip length would be as indicated by -4. *(See Figure 11.)* The range of application of last dash numbers is as follows:

-2 FROM .030 TO .077 (1/16-INCH)
-4 FROM .078 TO .140 (1/8-INCH)
-6 FROM .141 TO .203 (3/16-INCH)
-8 FROM .204 TO .265 (1/4-INCH)

16. DRILLING PREPARATIONS.

In drilling holes for the use of Cherry blind rivets, care should be taken to have the hole the correct size and at right angles to the sheets to be fastened together. The following finished hole sizes are recommended:

FOR 1/8-INCH RIVET	NO. 30 DRILL
FOR 5/32-INCH RIVET	NO. 20 DRILL
FOR 3/16-INCH RIVET	NO. 10 DRILL

Recommended finished hole sizes are shown plainly on each box of rivets as well as the limits of grip length. In special cases, these may vary from the standard given above and care should be taken to meet the dimensions given on each box.

17. DIMPLING PREPARATIONS.

In general, material for the use of countersunk Cherry blind rivets should be prepared in the same way as though solid rivets were to be used. In the use of countersunk Cherry blind rivets in dimpled holes, the usual steps must be taken to ensure proper straight-sided holes. Chips must be carefully eliminated from between the sheets, and any burrs resulting from drilling should be removed, as they prevent proper seating of the preformed head as well as the tulip-type head on the blind side of the work.

18. SHEET CLAMPING.

The use of skin fasteners and/or Clecos at reasonable intervals is recommended for holding sheets together before drilling or riveting. If skin fasteners and Clecos are not available, sheet metal screws may be substituted.

19. CHERRY RIVET INSERTION.

The hole sizes recommended in a foregoing paragraph are large enough for the rivet to slip in easily (see Figure 12). It is not necessary to have a snug fit. Careless handling of Cherry blind rivets sometimes results in nicks or dents on the thin edge of the preformed head. As this may prevent the head from seating properly, such rivets should be

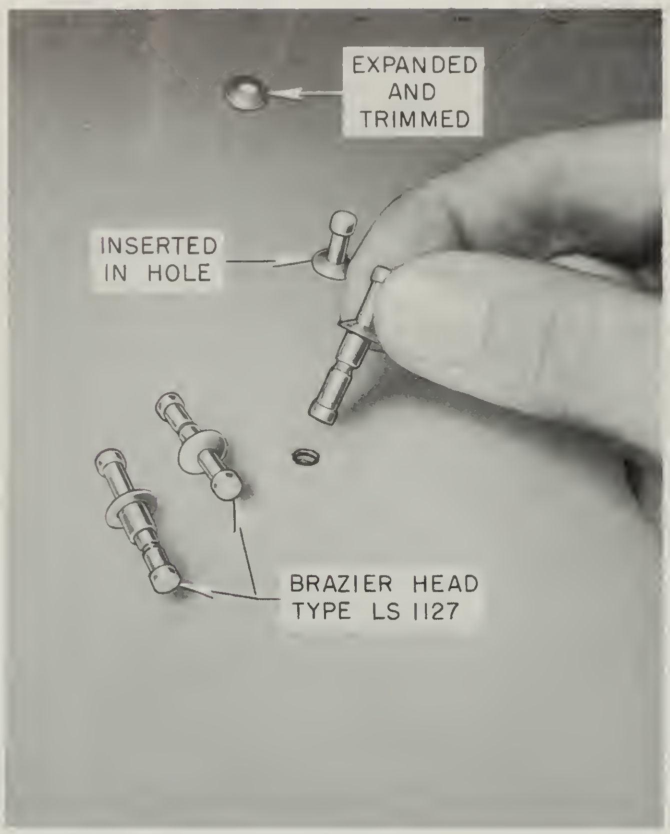


Figure 10—Inserting Cherry Rivets

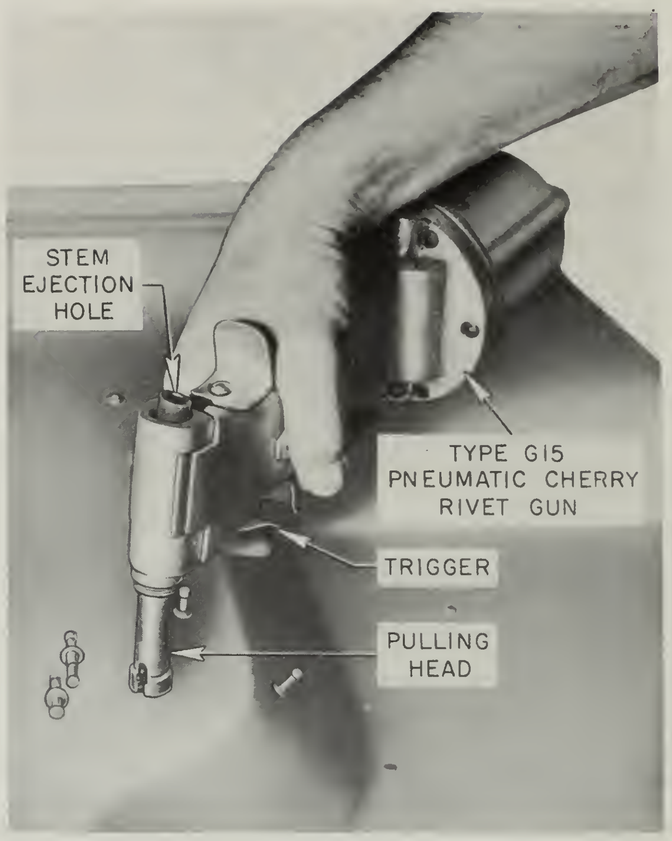


Figure 11—Expanding Cherry Rivets

discarded. Several rivets may be placed in their respective holes and then expanded by applying the rivet gun to the projecting stems. *(See Figure 10.)* In some cases insertion of the rivet into the gun head and then into the hole is preferable.

20. TYPES OF CHERRY RIVET GUNS

Two types of Cherry rivet guns are manufactured by the Cherry Rivet Company, a pneumatic and a hand type. These guns have interchangeable heads and can handle all sizes of Cherry rivets. Each head has two adapter caps, permitting the use of brazier head or countersunk type of rivets. The pulling heads on both guns operate on the same principle. Each has a slotted end into which the stem of the Cherry rivet is inserted. Upon the application of force, an inside draw bar pulls the head of the rivet against the gun and, by continued pull, forces the inside end of the stem into the tail of the rivet and eventually breaks the stem. *(See Figure 12.)* The G15 rivet gun, a pneumatic tool, operates on 75 to 90 lbs./sq.in. air pressure and exerts a 1300-pound pull on the rivet stems. The gun weighs 4 pounds and is balanced for easy handling at any angle. *(See Figure 11.)* The G10 gun is an improved, ratchet-type, hand-operated gun which can be used where air power is not available. *(See Figure 14.)* The ratchet feature permits the operator to exert an intermittent pull until the stem of the rivet breaks. A G6AA, 90-degree angle adapter is available for both types of gun. This adapter permits Cherry rivets to be inserted in locations where there is limited room for gun operation, such as between ribs in the wing structure.

21. USE OF CHERRY RIVET GUNS

Whenever applying a Cherry rivet gun on a rivet, make certain the rivet stem fits well into the bottom of the drawbolt slot before applying pressure. This should prevent shearing off the head of the rivet and ensure the breaking of the stem. Care should be taken to see that the head of the rivet is held firmly and squarely against the sheet to which it is being applied. This simple precaution will do much to ensure uniform and smooth results. The more nearly the axis of the pulling jaws is parallel to the axis of the rivet, the better the final job will be. Special instructions for the care and handling of each gun are delivered with the gun. Care should be used to see that proper nosepiece is fitted to the head of the

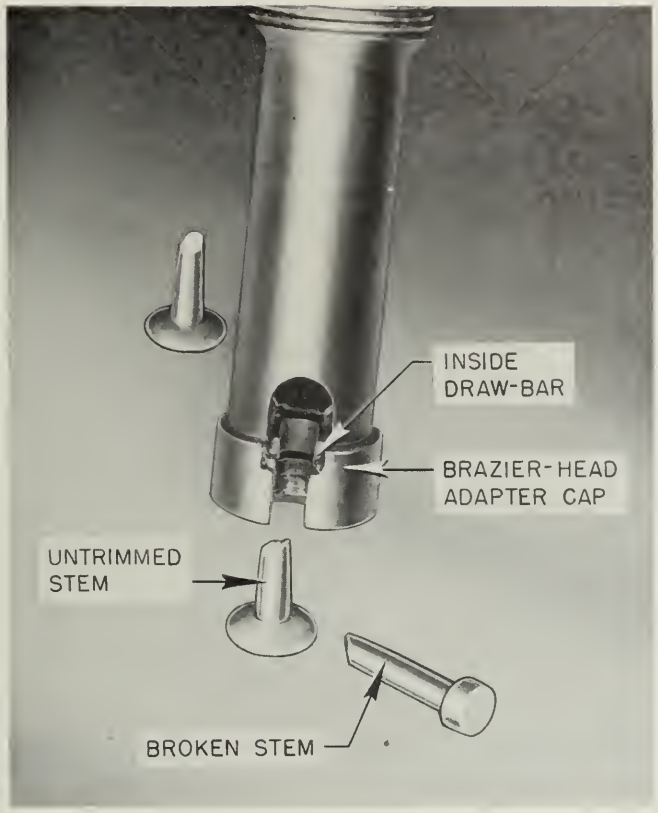


Figure 12—Untrimmed Cherry Rivets

Figure 13—Trimming Cherry Rivets

gun to handle either countersunk or brazier-type rivets in each diameter. It is necessary to have a separate size of pulling head for each diameter rivet. These heads are readily interchangeable.

22. CHERRY RIVET GUN MAINTENANCE

The maintenance and functioning of the gun are important. Care must always be taken to have the gun in proper working condition. This

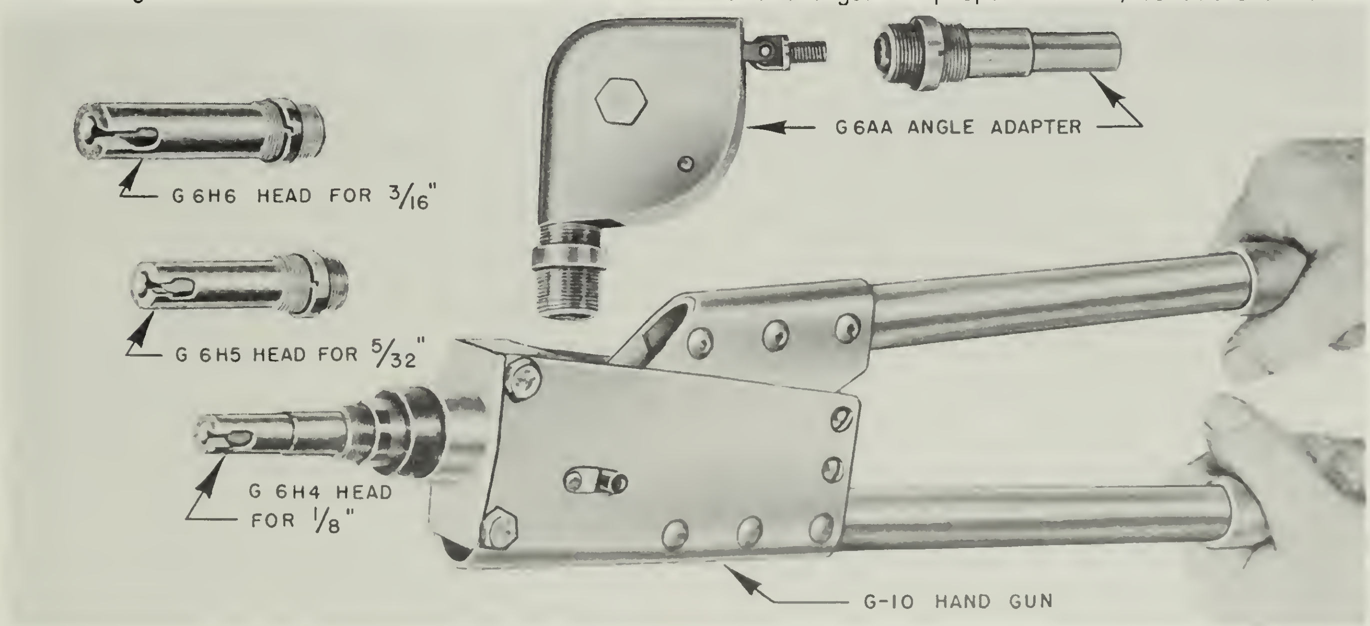


Figure 14—Cherry Rivet Hand Gun

can be determined by measuring the pulling head. In guns designed for self-plugging rivets, the movement is 3/4-inch minimum. Decrease in the length of the stroke may result in failure to break the stem of the rivet. Sometimes, instead of breaking the stem, the outside head will pull off. This is not a serious fault. Drill out the remaining portion and insert another rivet. When drilling out Cherry blind rivets, use a drill a size smaller than the diameter of the Cherry rivet stem. Punch the remaining portion out with a drift punch.

23. TRIMMING CHERRY BLIND RIVETS.

When the self-plugging rivet is applied, the stem usually breaks from 3/8-inch to 1/2-inch outside the preformed head. In most cases, it is desirable to cut off this stem. Such cutting can be done with an ordinary pair of multiplying nippers or diagonal cutters. *(See Figure 13.)* These should be ground down to a flat surface on the outside, and the inside should be hollow ground. If properly adjusted, such nippers will do satisfactory work.

DU PONT EXPLOSIVE RIVETS

24. GENERAL.

Du Pont explosive rivets, manufactured by E.I. du Pont de Nemours and Co., Inc., Wilmington, Delaware, present another solution to the problem of blind riveting encountered in field repair of aircraft. These explosive rivets are similar in appearance to standard solid rivets and the riveting preparations are the same. After insertion, a heated rivet expanding iron is applied to the head of the rivet and causes the rivet to explode and consequently expand on the blind side. This expansion forms a head on the blind side which grips the metal and prevents extraction of the rivet. Du Pont explosive rivets are suggested only as a substitution for solid rivets or Cherry blind rivets and should not be used indiscriminately.

25. RIVETING PRECAUTIONS.

Inasmuch as explosive rivets present a fire hazard when used on fueled airplanes, all necessary fire precautions must be taken. If the blind side of the riveting is in a closed compartment or bay, as will generally be the case, blow the compartment or bay with carbon dioxide or live steam to neutralize any vapors which might exist. Holes must be drilled carefully so there is a close fit between rivet and hole. Rivet of correct head type, shank diameter, and grip length must be used *(see Figure 15)*. The rivet must be expanded in 1.5 to 6 seconds with a recommended heating iron and a proper size tip. Parts to be riveted must be held firmly together at the time of rivet expansion. Do not use these rivets in steel or other metals harder than 24ST aluminum alloy. Do not attempt to expand rivets in any other way than with the standard du Pont riveting iron *(see Figure 16)*.

26. STORAGE AND HANDLING PRECAUTIONS.

Handle du Pont explosive rivets with care. Keep the rivets in the boxes in which they are

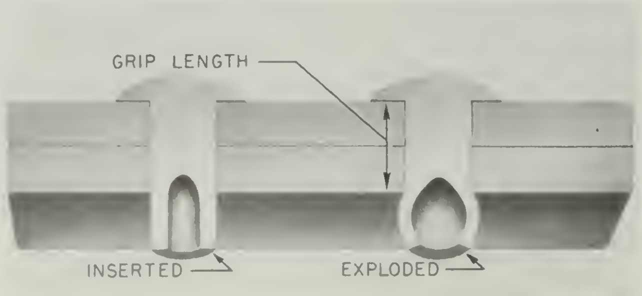


Figure 15—Explosive Rivet Cross Section

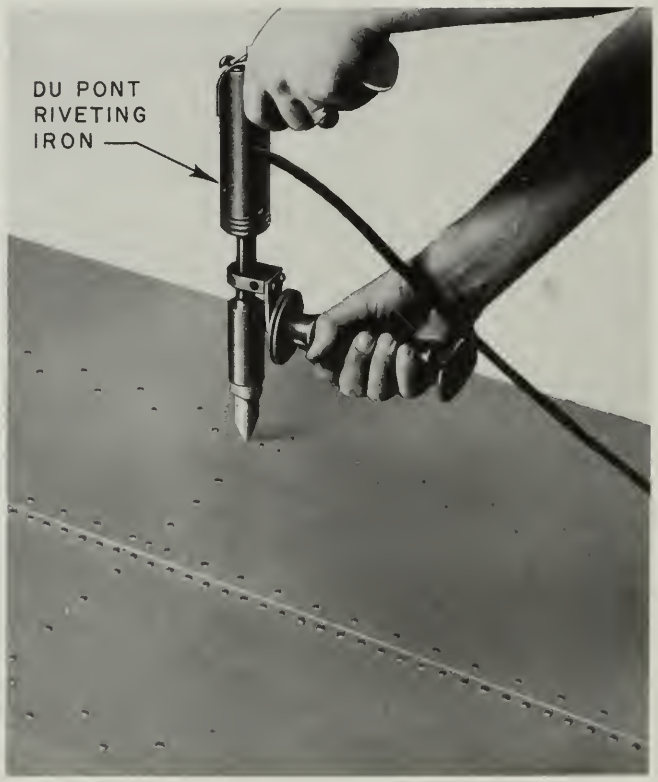


Figure 16—Use of Expanding Iron

received and store in a dry place, preferably in a closed cabinet. Keep open fires away and avoid temperatures in excess of 38°C (100°F). Do not throw unexpanded rivets into trash. If rivets become distorted so that they cannot be used, place them in a wire basket having holes smaller than the rivets and destroy them by placing the basket in a fire.

27. RIVET INSTALLATION.

Push rivet into drilled hole. Do not force to the extent of scraping the protective film on the shank. If necessary, tap lightly with a small wooden or rubber mallet, taking care not to damage the head of the rivet. Inspect to make certain the rivet head is well seated and the rivet shank fits snugly in hole. Make certain the rivet is of proper grip length and diameter *(see Figure 15)*. Apply a heated riveting iron to the head of the rivet, holding the tip firmly against the rivet head. Remove tip from rivet immediately after expansion.

Important: Du Pont explosive rivets should be expanded within 1.5 to 6 seconds after heat is applied.

28. DU PONT RIVETING IRONS.

The No. 3 iron is the most common riveting iron, for it can be used to expand all diameters and lengths of du Pont rivets *(see Figure 16)*. Six tips are available for this iron and it is important that the proper tip be used when expanding du Pont explosive rivets. Read carefully the instructions on the boxes in which they are shipped. The LOW setting on the wattage regulator is for short rivets. The HIGH setting on the wattage regulator is for long rivets. For intermediate length rivets, the regulator should operate at a point between LOW and HIGH in proportion to the length of the rivet. Try a sample in open sheet stock for proper expansion. The wattage regulator should be set so that the rivet will explode within 1.5 to 6 seconds after application of the expanding iron.

RIVNUTS

29. GENERAL.

A Goodrich rivnut is an internally threaded and counterbored tubular rivet that can be headed blind. Rivnuts are of a one-piece construction and are accurately machined from corrosion-resistant aluminum alloy. Rivnuts were designed primarily to be used as nut plates for the attachment of de-icers to the leading edges of wings and stabilizers. However, since rivnuts can be pulled up or headed while working entirely from one side of the structure, other unique adaptations have been made in secondary and primary structures of modern airplanes. The upset portion or head formed on the far side of the metal is large enough to resist being pulled through the metal skin even under conditions of eccentric load. After installation, rivnuts may be used as nut plates or as rivets, or both.

30. TYPES OF RIVNUTS.

Rivnuts are classified by the head style: namely, the countersunk and the flat head. The former is made in three different head shapes while the latter is made in only one. Each style is made in three sizes, No. 6-32, No. 8-32, and No. 10-32. Each of these sizes is available in six grip ranges and can be furnished with open or closed ends. With the exception of the thin-head countersunk type, all styles can be furnished with or without keys under the head. The closed end rivnuts should be used only in special places such as sealed compartments for flotation bays. Keyed-type rivnuts should be used whenever screws are to be inserted to increase strength or for the attachment of fittings.

31. GRIP RANGE.

In order to produce an ideal bulge or head on the blind side of the metal sheet, a positive relationship must be maintained between the counterbore depth of a rivnut and the thickness of the material in which it is being headed. This thickness is known as the grip. The grip range is the thickness of the material above and below this ideal point in which a rivnut may be successfully headed. Six consecutive and nonoverlapping grip ranges are available and these grip ranges are identified by indented markings found on the exposed surface of the head. Absence of these marks classifies the rivnut in the minimum range of its head style. One mark indicates the next higher range and five marks indicate the maximum range. The grip range of any rivnut can be determined directly from its type number, as the figures at the right in the rivnut type number indicate the maximum grip in thousandths of an inch and the minimum grip for this rivnut is the same as

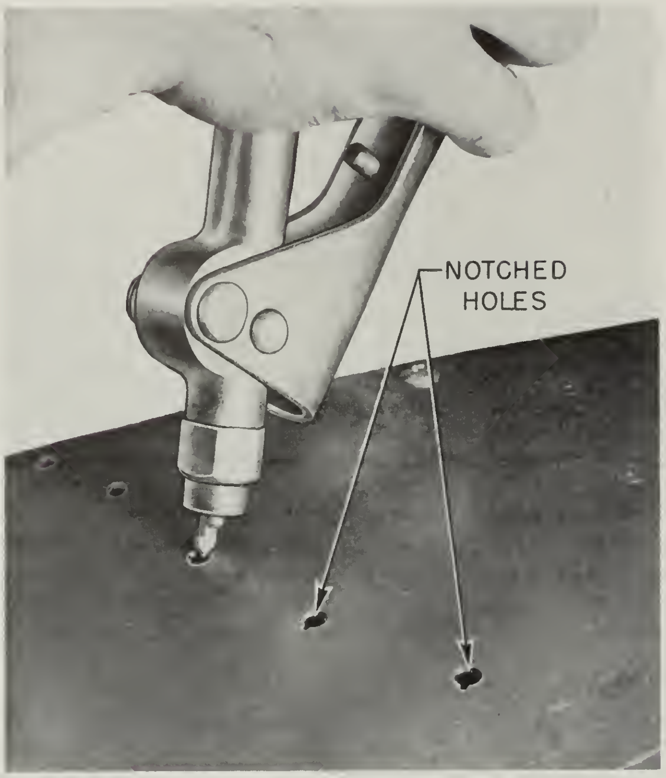


Figure 17—Use of Keyseating Tool

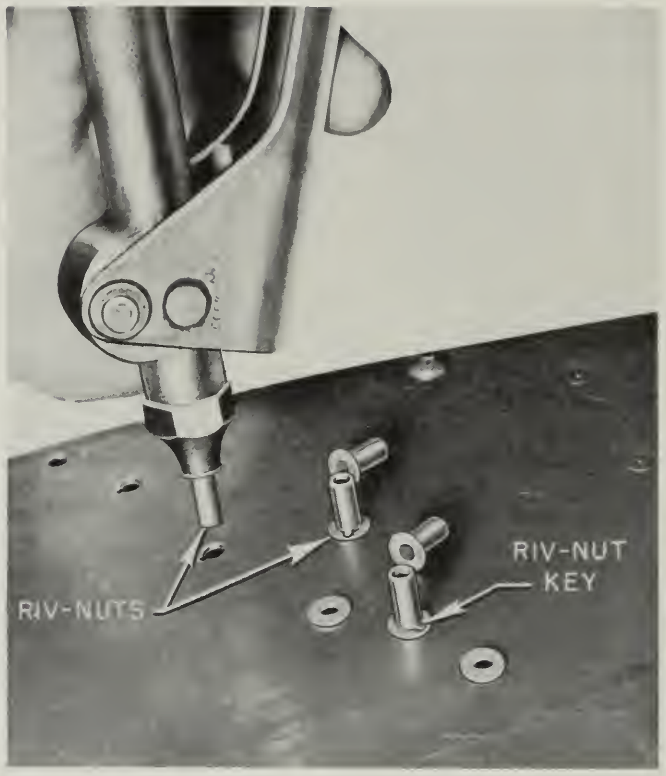


Figure 18—Inserting Rivnut

the maximum grip of the preceding size, except in the case of the first rivnut in a series, where the minimum grip is the thickness of the head in the countersunk style and .010 inch in the flat head style. For best results, a rivnut should be used only within the grip range for which it is intended.

32. RIVNUT TYPE NUMBERS.

The first figure of a rivnut number indicates the screw size of the internal thread. The figure at the right indicates the maximum grip in thousandths of an inch. The minimum grip equals the maximum grip of the preceding size, except that the minimum grip of the first rivnut in a series is the head thickness for the countersunk types and .010 inch for flat head types.

A dash between figures indicates OPEN END KEYLESS
A "B" between figures indicates CLOSED END KEYLESS
A "K" between figures indicates OPEN END WITH KEY
A "KB" between figures indicates CLOSED END WITH KEY

If the figures at the right are evenly divisible by 5, the rivnut is the flat head type. If the number is not evenly divisible by 5, the rivnut is a countersunk type.

Examples: An 8KB100 rivnut is an 8-32, CLOSED END WITH KEY rivnut and has a maximum grip of 100 thousandths of an inch (.100") and a minimum grip of 75 thousandths of an inch (.075"). A 6B106 rivnut is a 6-32, CLOSED END KEYLESS rivnut and has a maximum grip of 106 thousandths of an inch (.106") and a minimum grip of 63 thousandths of an inch (.063"), which is the countersunk head thickness. The right-hand number (106) is not evenly divisible by 5. Therefore the rivnut is of the countersunk type.

33. STRENGTH OF RIVNUTS.

Before substituting rivnuts for solid rivets in all repair work, a comparison of the shear and the tensile strength must be made to ensure a resultant safe repair. The shear and the tensile strength of rivnuts may be greatly increased by the insertion of standard screws into the hollow stem. When screws are omitted, the shear strength of rivnuts is approximately 30 to 50 percent of that of solid rivets. The insertion of standard SAE 2330 screws increases

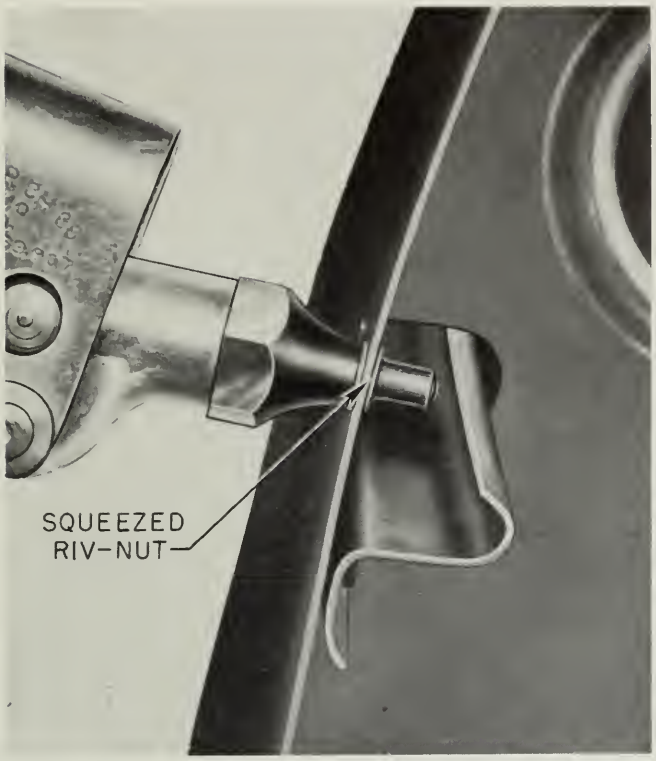


Figure 19—Squeezing Rivnut

Figure 20—Removing Mandrel

the strength considerably. Before substituting rivnuts for solid rivets, consult a qualified engineer. The ultimate shear and tensile strength of solid rivets set forth in the ANC-5 (Strength of Aircraft Elements) Manual should be compared with the shear and tensile strength set forth by the Goodrich Rivnut Company, to determine the size and number of the rivnuts to be used.

34. PREPARATIONS FOR INSTALLATION.

In general, material for the use of rivnuts should be prepared in the same way as though solid rivets were to be used. As much precision is required in drilling a hole for a rivnut as for a regular rivet, since the shank of the rivnut must fit snugly into the hole. In order to obtain a smooth round hole, a lead hole smaller in diameter than the body size of the rivet should be drilled first and followed by the correct finished hole size.

SIZE RIVNUT	LEAD DRILL	NO. BODY DRILL
NO. 6-32	NO. 19 (.166")	NO. 12 (.189")
NO. 8-32	NO. 8 (.199")	NO. 2 (..221")
NO. 10-32	NO. 1 (.228")	¼" (.250")

When countersunk rivnuts are to be used in cut countersunk or dimpled holes, the usual steps must be taken to ensure proper straight-sided holes. Chips must be carefully eliminated from between the sheets, and any burrs resulting from drilling should be removed, as they prevent proper seating of the preformed head as well as the upset head on the blind side of the work. The use of skin fasteners at reasonable intervals is recommended for holding the sheets together before drilling, notching, or upsetting operations are accomplished. If skin fasteners are not available, sheet metal screws may sometimes suffice.

35. RIVNUT INSTALLATION.

After the hole has been properly drilled, it must be notched to permit the proper seating of the key. Insert a keyseating or notching tool into the hole and hold it firmly against the sheet. *(See Figure 17.)* Squeeze the handles of the notching tool together. This action cuts a notch, or keyway, in the side of the hole. When the thickness of the material prevents the use of this keyseating tool, the keyway may be formed with a small round swiss needle file. All heading tools have a threaded mandrel on which the rivnut is to be threaded until the head of the rivnut is against the anvil of the heading tool. The rivnut is then inserted in the hole and the key positioned with respect to the keyway. *(See Figure 18.)* Then, while holding the tool so the mandrel is at 90 degrees to the surface of the metal, the mandrel is retracted by squeezing the handles together. This draws the rivnut against the anvil, thus causing the bulge or head to form in the counterbored portion of the rivnut on the inaccessible side of the work. *(See Figure 19.)* Squeeze the handles together until solid resistance is felt. Excessive pressure beyond this point is unnecessary. Turn the handwheel counterclockwise to unthread and withdraw the mandrel from the installed rivnut. *(See Figure 20.)* Hand-type tools are illustrated. Pneumatic power rivnut drivers are available and threading, upsetting, and withdrawal are accomplished by compressed air through the manipulation of finger-tip controls.

DZUS FASTENERS

36. GENERAL.

Dzus fasteners which are broken or damaged must be replaced with new ones *(see Figure 21)*. The light-duty type which is found on box covers, access hole coverplates, and fairing sections are easily removed and replaced. The heavy-duty type, which is generally used on cowling and heavy fairings, requires two additional operations, inasmuch as the retaining grommet must also be removed and replaced.

37. REMOVING LIGHT-DUTY DZUS FASTENERS.

The removal and replacement of light-duty dzus fasteners are accomplished with the use of a set of TJ 1033A punch and die dzus fastener tools or facsimiles. Center the broken dzus fastener over the hole in the anvil with the grooved end up. Place the proper-sized drift punch squarely on the end of the fastener. Hold the punch firmly in place and strike the punch a sharp blow with a hammer *(see Figure 22)*. After the broken or damaged dzus fastener is driven out in this manner, the hole will be too large and will be dimpled slightly on the wrong side. Remedy this condition by flattening the hole area with the flat end of a ball-peen hammer and an anvil *(see Figure 22)*.

38. REPLACING LIGHT-DUTY DZUS FASTENERS.

After the broken fastener has been driven out and the hole area flattened, as set forth in the preceding paragraph, the hole must be redimpled

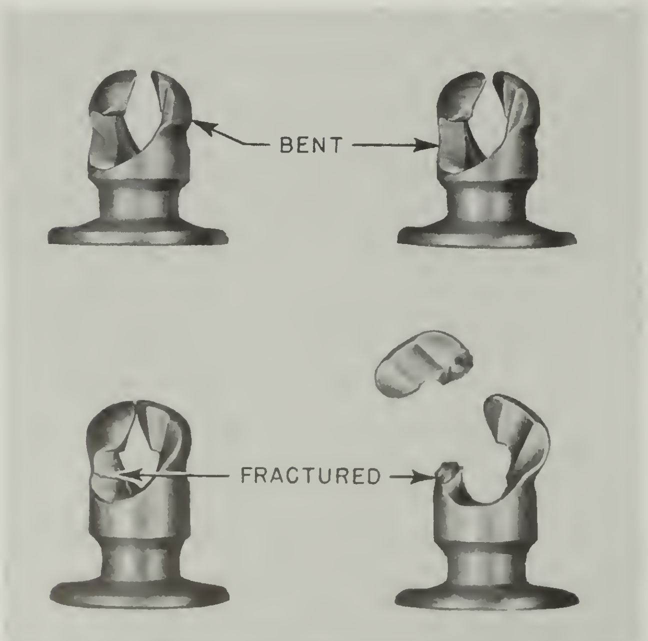


Figure 21—Typical Damage to Dzus Fasteners

Figure 22—Dzus Fastener Tool

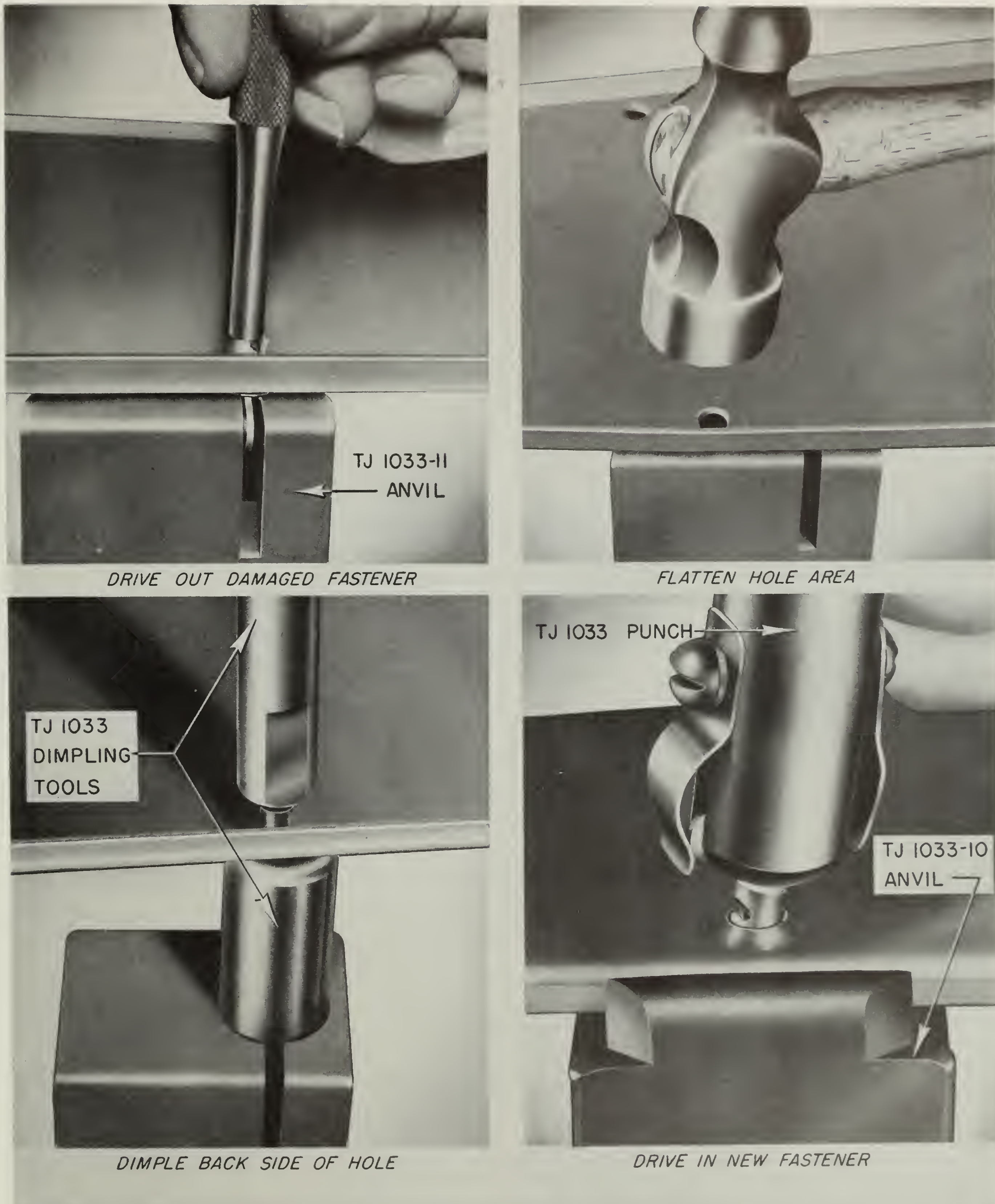


Figure 23—Light-duty Dzus Fastener Replacement

with a proper size and degree dimpling tool *(see Figure 23)*. After the hole has been dimpled, turn the lid or coverplate over and fit it on the anvil so the dimpled hole is directly centered over the hole in the anvil. Insert a new fastener into the special driving tool, insert the grooved end of the fastener into the dimpled hole, and drive into place with a hammer *(see Figure 23)*.

39. REMOVING HEAVY-DUTY DZUS FASTENERS.

To remove damaged heavy-duty dzus fasteners, it is necessary first to remove the retaining grommet. A modified hole saw with no pilot may be used to accomplish this *(see Figure 24)*. After the grommet has been cut as much as possible, it may be nipped off with a pair of diagonal cutters. New grommets and fasteners may be obtained from the Dzus Fastener Co., Inc., Babylon, N.Y.

40. INSERTING NEW FASTENER.

Insert a new grommet into the hole in the cowling or fairing and expand the small end with a tapered punch *(see Figure 25)*. The grommet must be dimpled to such an extent that it will not fall out of the hole, but not to such an extent that the new fastener cannot be inserted. Insert the new fastener into the retaining grommet and place over a hollow anvil. With a square-ended driving tool, drive the new fastener down into the retaining grommet until the grommet is flattened and securely grips the surrounding material *(see Figure 25)*. A special dzus fastener tool, TJ 4451, is available as an aid in fastening flush-type dzus fasteners such as found on cowling and large fairings. This tool has three blades, each of a different thickness to fit the various grooves found in flush-type dzus fasteners *(see Figure 22)*.

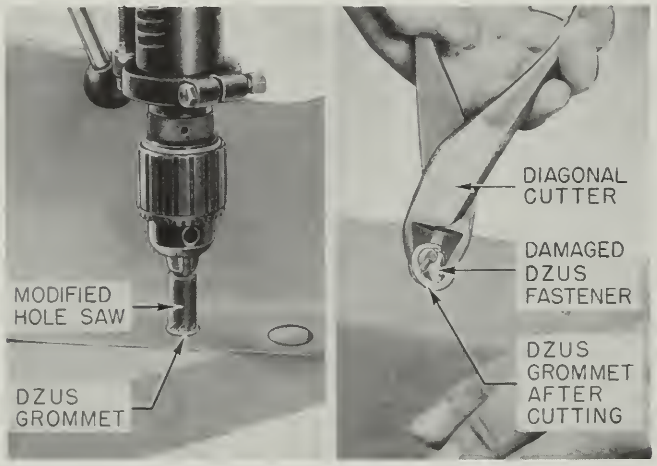


Figure 24—Removing Grommet

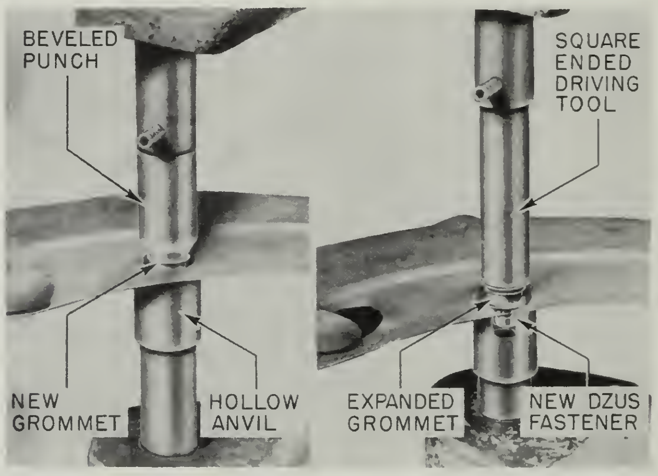


Figure 25—Inserting Grommet and Fasteners

TRANSPARENT PLASTIC PANELS

41. CLEANING.

In order to avoid scratching the panels, it is recommended that the panels be washed with a grit-free soft cloth, chamois, or sponge. In addition to scratching the panels, wiping with a dry cloth builds up an electrostatic charge on the panels which attracts dust particles. Washing with a damp chamois will remove this charge and the dust and will not scratch the surface. Soap and water or carbon tetrachloride may be used to remove grease and oil; but acetone, benzine, and lacquer thinners will affect the surface. The application of a wax finish is recommended to protect the panels from undue scratching.

42. MINOR SCRATCHES.

Minor scratches may be removed by vigorously rubbing the affected area by hand, using a soft cloth moistened with a mixture of turpentine and chalk or automobile body cleaner. Rub the area both in the direction of the crack and at right angles to the crack; but do not rub too long in one place, as the heat generated by friction may cause ridges. When the scratches are removed or considerably improved, apply a final wax polish.

43. DEEP SCRATCHES.

If the scratches are too deep to be removed

Figure 26—Removing Deep Scratches in Transparent Plastic Panels

by the procedure outlined for minor scratches, it may be necessary to sand the area around the scratches. It is to be noted, however, that panels should not be sanded unless absolutely necessary. Wrap a sheet of 320A fine sandpaper around a block, wet the sandpaper, and sand lightly over a wide area with a free circular motion *(see Figure 26)*. Do not confine the sanding to too small an area, or objectionable distortions or "bull's-eyes" will result in the transparent plastic. Wash the surface; then sand lightly with a wet sheet of 400A or finer sandpaper. When the primary deep scratches are removed, remove the fine sandpaper scratches by the procedure outlined in the paragraph on Minor Scratches. If a buffing wheel is available, a more satisfactory method is to apply a preliminary polish to the deep scratches by means of a felt disc coated with a mixture of jeweler's rouge and water. Rotate the disc at approximately 250 RPM. Apply lightly and keep moist. After the depth of the scratches has been reduced, apply a final polish or turpentine and chalk with an 8-to 10-inch diameter silk buffing wheel rotated approximately 2000 RPM. Clean the area and apply a wax coat.

44. REPAIRS - GENERAL.

The repairs which could be attempted in the field are at best makeshifts and should not be regarded in any other light. Although it is intended to suggest methods which could be followed in the shortest possible time and with the simplest possible equipment, obviously any damaged pieces should be replaced as soon as possible. Even when adequate repair facilities are available, replacement of the part may be less expensive than attempting a repair. In general, it will be found economical to buff almost any scratched section, but patching may not be justified except on large and complicated formed sections. Besides the considerations of cost, it should be borne in mind that even a carefully patched part is not optically or structurally equal to a new section.

45. ISOLATED CRACKS.

At the first sign of cracking, a hole 1/8- to 3/16-inch in diameter should be drilled at the end of the crack. This simple operation helps to prevent further splitting by distributing the strain over a larger area. If the crack is small and repair facilities limited, stopping the crack with the drilled hole usually will suffice until a more permanent repair or a replacement can be made. However, if more permanent repair is to be effected, plug the drill hole as outlined in the following paragraph, apply Lucite Cement, H-65, (E. I. du Pont de Nemours & Co., Inc.) with a brush to both surfaces of the crack, and then to the crack apply a stream of intense hot air with a blower *(see Figure 27)*. Play the blower over the crack with a circular motion for approximately 5 minutes; then let the material cool. If a hot-air blower

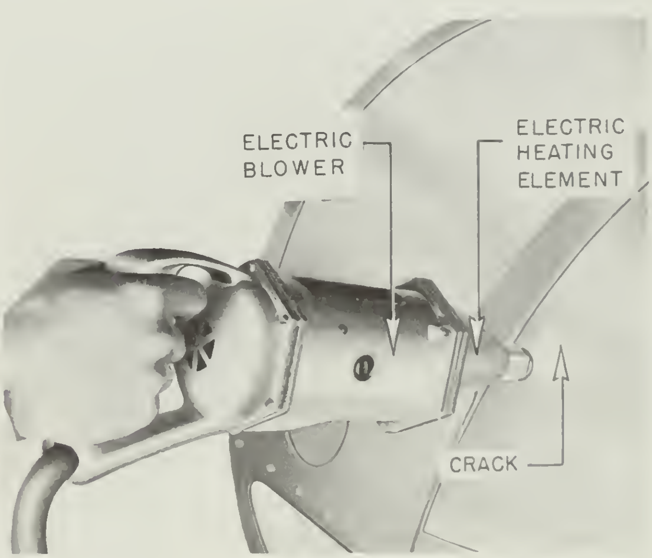


Figure 27—Applying Hot-air Blast to Cracked Plastic Panel

Figure 28—Applying Torch Flame to a Cracked Transparent Plastic Panel

is not available, a torch may be substituted. The torch should produce a soft bluish flame and should not be blowy. Hold the torch several inches from the panel *(see Figure 28)*. This type of repair may be regarded as permanent.

46. REPAIRING HOLES.

The first step in repairing holes in transparent plastic panels is to trim the hole and surrounding cracks to a circle as soon as possible. This will prevent the development of radial cracks and will confine the damage to a minimum area.

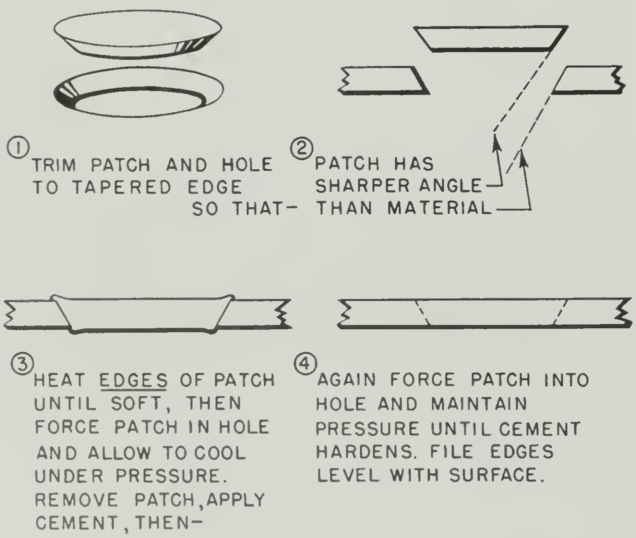


Figure 29—Patching Holes in Transparent Plastic Panels

Trim the edges of the hole to a 45-degree taper; then cut a transparent plastic patch of slightly thicker material than the section to be repaired and trim its edges to a sharper angle than the hole *(see Figure 29)*. Heat the edges of the patch over a hot-air blower or an alcohol lamp until the edges are very soft and pliable; then force the patch into the hole and allow it to cool. Remove the patch and soak the edges of the patch in Lucite Cement, H-65, (E.I. du Pont de Nemours & Co., Inc.) for 2 or 3 minutes; then insert the patch into the hole again. Because the edges are tapered, pressure need be applied only on one surface and equal pressure will automatically be applied to all edges. Leave the patch under light pressure for upwards of 24 hours; then after the cement has hardened, sand or file the edges level with the adjacent surface. In patching a curved or a large section where it is not possible to heat the patch, it is possible to obtain a perfect fit by cutting the patch first and using it as a template.

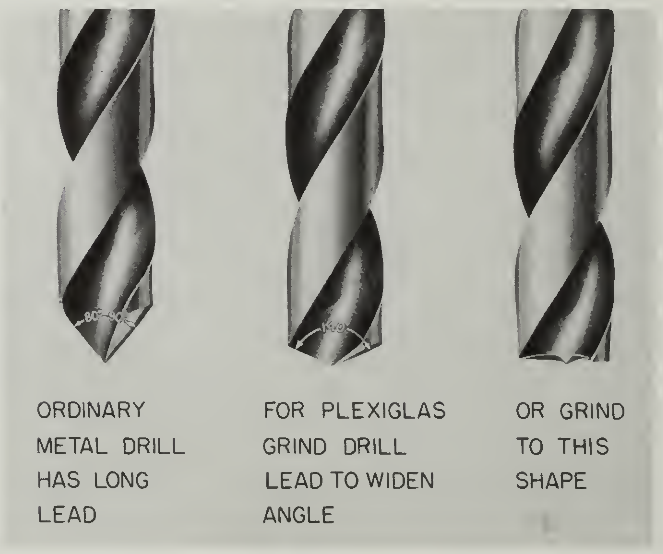


Figure 30—Modified Drills for Transparent Plastic Panels

The area to be cut out of the damaged piece is indicated by scribe lines. The operator should saw within the scribe lines and sand or file the edges to a smooth 45-degree taper. With careful workmanship, it will be possible to obtain a good fit. The patch should always be oval or round in shape, never square.

47. PREPARATION OF NEW CURVED PANELS.

Where spare curved panels are available, they should be utilized as replacements. However, where spare panels are not available, replacements may be prepared from flat stock. The material should consist of transparent acrylate-base plastic sheet of a thickness equal to the original panel. In order to provide sufficient plasticity for working, hang the trimmed flat panel in a hot-air oven at 104°+121°C (220°+250°F). If the hot-air oven is not available, the flat panel may be heated in hot oil, glycerin, or kerosene. Hot water is unsatisfactory and should not be used. If the heated liquid method is used, immerse the panel for a few minutes, remove it, and allow it to form over a cloth-covered wood or metal form block. Since the warm plastic readily takes impressions, it must be handled carefully with clean cotton gloves. After the plastic sheet has cooled on the form, remove, trim, and polish the panel. To drill the panel for installation, ordinary drills for metal may be used. Better results are obtained, however, if the drills are ground with very little lead *(see Figure 30)*. To prevent clogging and burning of the sheet, lift the drill from the hole frequently and remove shavings. Use moderate speeds and light pressure to avoid "grabbing" when the drill penetrates the transparent plastic. If high speeds are used, a coolant may be required. If the point of the drill penetrates thin material before the full width has entered, the drill will tear the sheet. The fabricated panel must be installed exactly like the original.

48. ROUTING EDGES OF PANELS.

After a new panel has been cut and formed to fit an enclosure, the edges must sometimes be

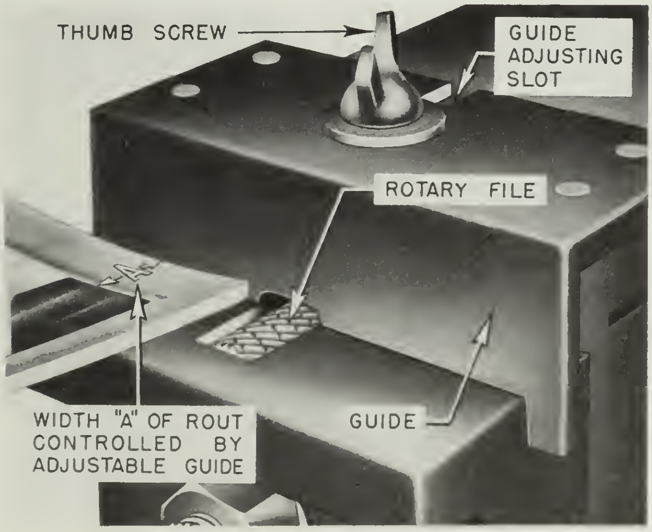


Figure 31—Panel Routing Jig

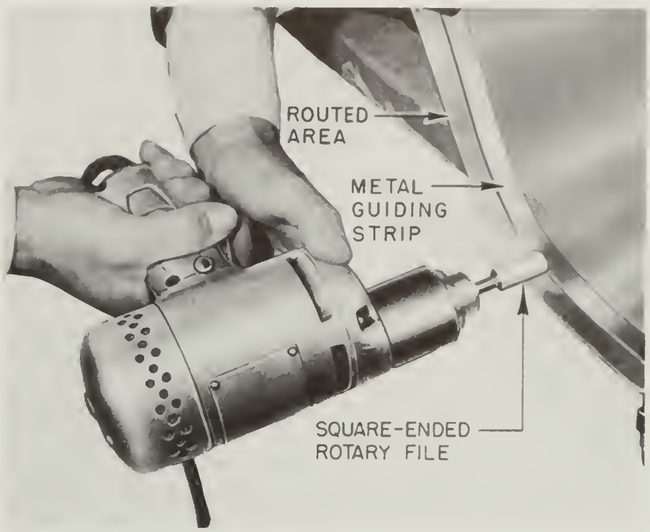


Figure 32—Routing Inside Curves

routed or skived. This may be accomplished by several methods, and the method used will necessarily be dependent upon available equipment. Two practical ways, accomplished with simple tools generally found in any shop, are described in this and the following paragraph. The tools required are a square-ended rotary file of at least 1/2-inch diameter and any means of motive power capable of spinning the rotary file at a speed of at least 2000 RPM. By means of metal blocks and a piece of angle iron, form a routing jig *(see Figure 31)*. Cut a slot in one flange of the angle iron and insert a thumbscrew into this slot. By means of this adjustable guide the width of the cut can be accurately controlled. Some means must also be provided to control the depth of cut by the rotary file. This must be determined locally, as the method of applying motive power to the rotary file will also be a local problem. An air supply nozzle at the cutting blade is suggested to keep the rotary file from clogging. Place the edge of the panel against the guide and pass the panel over the cutting blade. If a deep cut is to be taken, better results will be obtained if it is accomplished in two or more operations. This jig method will satisfactorily rout the edges of straight panels or the outside periphery of curved panels.

49. ROUTING INSIDE CURVES.

Inasmuch as the method of routing transparent plastic panels described in the preceding paragraph cannot be applied to the inner surface of a curved panel, a method such as illustrated must be used. *(See Figure 32.)* Obtain a narrow strip of soft metal and form it to fit the contour

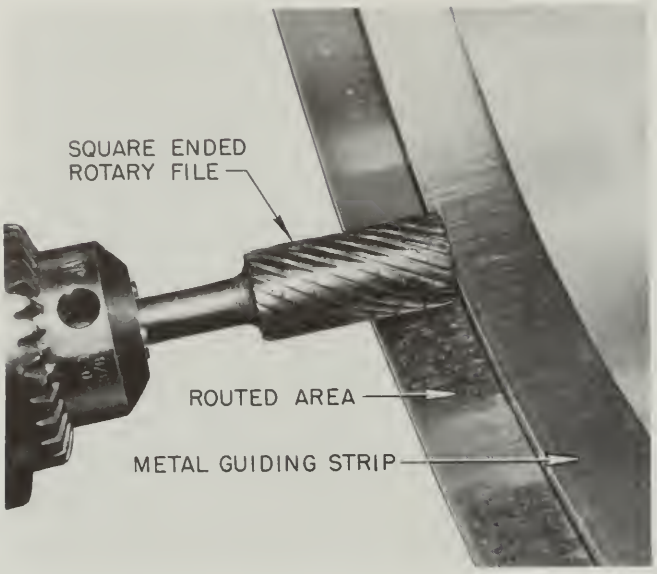


Figure 33--Curve Routing Detail

panel. With small "C" clamps or other suitable means, clamp this metal strip accurately into position along the edge of the panel, exposing only as much of the panel edge as is desired to rout *(see Figure 33)*. When this method of routing is applied, the motor which turns the rotary file must be held by hand and the depth of the cut cannot be accurately controlled. It is suggested that a solid rest be provided for the motor to give added control. Pass the rotary file back and forth over the exposed area, keeping the flat end of the rotary file against the metal guiding strip *(see Figure 33)*. An alternate method can be used when the panels are small. Clamp the motor securely in a vise and pass the exposed portion of the panel back and forth over the rotary file, keeping the flat end of the rotary file against the metal guiding strip.

50. COCKPIT ROOF PANEL REPLACEMENT.

A visual inspection of the cockpit roof will readily determine the procedure for the removal and replacement of individual panels. To remove a panel from the windshield assembly, remove the bow and strips as one assembly by drilling out the rivets and removing the screws securing the assembly to the cowling. Disassemble the frame as required and replace the defective panel. If the rubber strips and the sponge rubber located in the grooves of the frame are in good condition, they may be used again. Apply a plastic caulking compound around the outside edges of the windshield, such as Hunt's Mastic Caulking Compound or the equivalent.

51. SIDE PANEL REPLACEMENT.

To replace any of the side panels in either

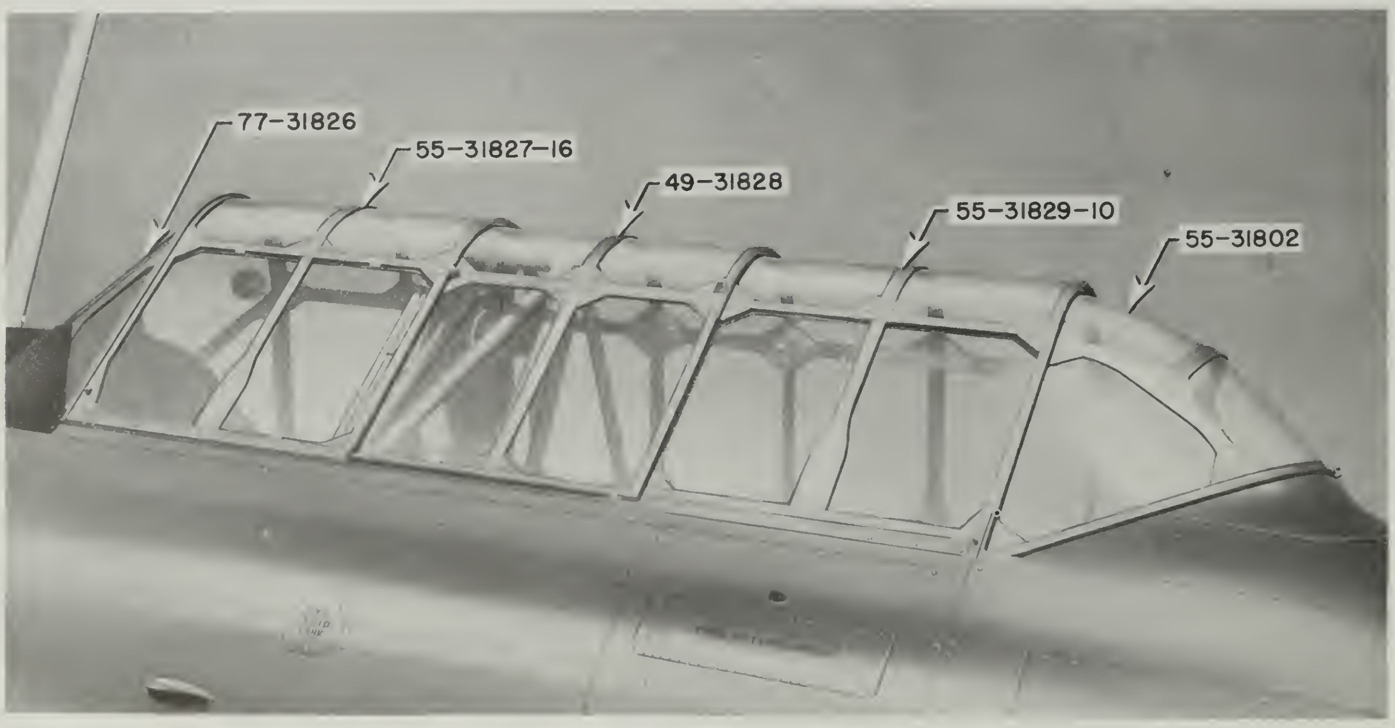


Figure 34--Cockpit Enclosure Assembly

of the curved sliding section, pull the EMERGENCY handle and push the defective panels out with the hand. Replace with the new panel, utilizing the same rubber strips if in good condition. Ascertain that the EMERGENCY handle lockpins and safety wires are properly reinstalled. To replace a forward top panel of the aft sliding section, removal of the entire rear hood assembly is necessary; this is to permit the sliding section to be moved aft. Remove the bow and corner blocks if in good condition. To remove the aft top panel of the section, it is only necessary to remove the bow and corner blocks. To replace a forward side panel or top panel of the stationary section, remove the four block-shaped stops from the inside of both tracks. Lift the entire section clear. Remove the bow and corner blocks and replace the defective panels as required. To remove an aft top panel, it is only necessary to remove the bow and corner blocks. When ordering a replacement part, be sure to give the correct part number. Each panel has a separate part number and is called out on its respective assembly number *(see Figure 34)*.

ALUMINUM CASTINGS AND FORGINGS

52. GENERAL.

The repair of castings and forgings by welding is not recommended. Damaged parts should be replaced by spares. However, minor repair by welding of parts not requiring high structural values is permissible, provided that special practices inherent in the process be observed. It should be borne in mind that, in the majority of cases, the use of castings and forgings is utilized as a means of expediting production; therefore, as a means of temporary repair, a substitute part can be fabricated of some other material and of comparable design. Where fabrication of parts is accomplished, the ultimate strength and functional requirements of the original must be maintained. See Section XII for the required tensile strength of all castings and forgings subsequent to heat treatment, and Section I for possible corresponding material substitution equivalents.

53. MINOR SURFACE CRACKS.

Small surface cracks, scratches, and minor discontinuities may be relieved by carefully filing or scraping. Filing should be such as to present a gradual, smooth filleting of the area affected. Sharp or abrupt corners should be avoided. The area should be thoroughly cleaned and a clear varnish or other transparent solution suitable for the purpose should be applied as an aid in determining any subsequent development of the discontinuity.

54. MINOR WELDING OF LOWLY STRESSED MEMBERS.

In general, welding of castings and forgings varies little in practice from that of sheet material, except that necessary allowances for variations in shape, cross-sectional area, and form must be made. Prior to welding a casting, carefully investigate the nature of construction. Look for tapering cross sections which would tend to draw the welding heat away and make preheating of the casting necessary. Observe the possibilities of thermal strains and cracks, which are most likely to occur in castings of thin-walled or otherwise intricate design. After the proper investigation has been made, thoroughly clean the area by means of a stiff-bristled wire brush and gasoline. If the damage is a minor crack, completely cut or melt away the crack and a portion of the surrounding stock by means of a file or puddling iron. If a small loose piece is to be replaced in the casting, carefully clamp the piece in position, securing loosely to prevent the thermal strains during welding. Larger castings or castings having intricate sections should be preheated by means of a suitable furnace prior to welding. Small castings or castings requiring a weld near the edge of a thin-walled section may be preheated locally by means of a welding torch. In either case, preheating must be accomplished gradually and at uniform rate. Use a suitable welding flux and a welding rod of the same material as the casting. The added material must be completely molten and must be thoroughly worked with the end of the welding rod. Remove accumulated oxides and impurities from the surface of weld by means of a wire brush. After welding, heat treat the casting or forging in accordance with the required tensile strength specified in Section XII.

Figure 35—Propeller Installed

55. MINOR REPAIRS.

Visually inspect the hub and blades for damages such as nicks and sharp dents on the leading edges, or gashes on the blade faces. Such injuries are particularly dangerous, as they greatly reduce the fatigue strength at that particular point. A failure may result unless they are promptly removed. All mars on the surfaces of the blades are "stress raisers" and cause a stress concentration which may raise the stress beyond the endurance limit, resulting in a fatigue failure. Sharp dents and nicks or gashes may be removed locally without the necessity for reworking the entire blade surface. A curved "riffle" file is recommended for use in removing the sharp base of the nick. Fine emery cloth or crocus should be used for polishing. Care should be taken in removing nicks from the blade face to ensure that the thickness is not reduced more than is necessary. It is recommended, as an added safety precaution, that the surface be etched after the removal of a nick and examined with a magnifying glass to ensure that the nick is entirely removed and that a crack has not started. After inspection, polish the etched area. Propellers having very severe nicks or gashes should be sent to an authorized repair depot or returned to the factory for repair. *(See Figure 35.)*

WOODEN ACCESSORIES

56. WOODEN COCKPIT SEATS.

Damages to the wooden cockpit seats may be repaired by following the procedures set forth in Section II for preparing and gluing wooden splice members and patches to plywood panels. Inasmuch as the seats are a nonstructural part of the airplane, the scope of negligible damage is much greater than that of a structure member. However, it is essential to have no rough edges or protruding members attendant after repair. No repairs should be made that will interfere with, or result in damage to, the seat harness, the safety belt, or the seat adjusting mechanism. *(See Figure 36.)*

57. WOODEN FLOOR BOARDS.

General methods set forth in Section II for gluing, clamping and reinforcing damaged wooden

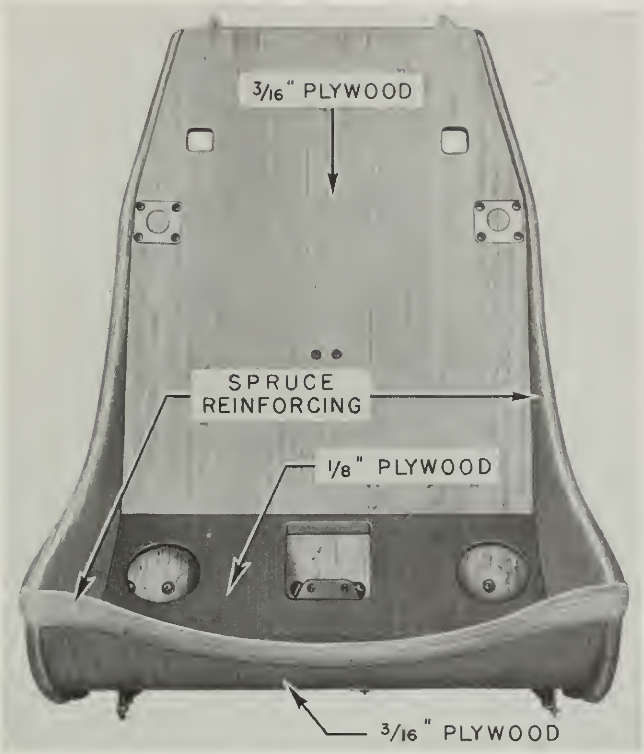


Figure 36—Wooden Cockpit Seat

members may be used to repair the wooden floor boards of the front and rear cockpits on the later model airplanes. All plywood used for repairs should conform to AAF Spec. AN-NN-P511A. Spruce used for reinforcing and splicing members should conform to AAF Spec. AN-S-6a. Casein glue conforming to AAF Spec. C-G-456 should be used for gluing all repair members to the original parts. Attach all repair members to the underneath side of the floor boards so as not to interfere with any movements of the pilot's feet.

58. MINOR WOODEN ACCESSORIES.

Other miscellaneous small items on the airplane are constructed of plywood and reinforced with spruce. The majority of these items are considered as expendables and their size and minor cost warrants replacement rather than repair. All such items may be purchased from North American Aviation Inc., Inglewood, California. If repair is attempted, the primary consideration must be the function of the part after repair. No protruding members should be attendant after repair and the strength of the repaired part must equal that of the original.

CONTROL RODS

59. GENERAL.

Various units of the engine control installation and the fuel and oil system installations are controlled by resistance welded push rods or, as in the case of the fuel selector valve, by torque rods. Repairs to these control rods consist of smoothing out negligible damage, splicing damaged rods, manufacturing replacement rods, and replacing the clevis, threaded, or ball-bearing rod ends. The repair must develop the original strength of the rod and be able to withstand all the compression, tension, or torque loads imposed upon it. All splices must be so made as not to interfere with the movement of any of the controls, and special attention must be paid to locations where control rods pass through the firewall or through phenolic fiber fairleads. No welds are permitted within a region on the control rod that will pass through a fairlead.

60. NEGLIGIBLE DAMAGE.

Smooth dents not exceeding one tenth the tube diameter or one tenth the tube perimeter, which are without cracks, fractures, or sharp corners, may be disregarded except to satisfy appearance. Smooth the dent with a half-round fine file and finish with steel wool. Apply the original finish to the member. *(See Section XI.)*

61. SPLICING CONTROL RODS.

Severely damaged or badly bent control rods

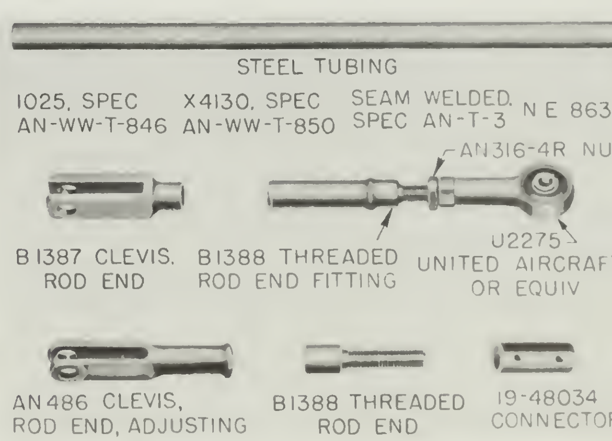


Figure 37—Control Rod Repair Components

may be spliced by welding in a section of steel tubing to match that of the original rod. Steel tubing conforming to four different specifications may be used for the splice member. *(See Figure 37.)* Proceed as follows: Cut the rod on each side of the damage. Obtain a length of steel tubing to match the diameter and wall thickness of the damaged rod, and cut to such a length that the length of the rod when spliced will equal the length of the original rod. Clean and burr all rod ends and file to a 60-degree taper. Drill a No. 40 hole in the tube to provide for the injection of the corrosion inhibiting fluid. Butt the rod ends together and carefully align the splice member with the original rod. With suitable clamping devices, hold the rods firmly in place. With oxyacetylene welding equipment, apply a weld around the joint, completely filling the beveled gap. *(See Figure 38.)* For all information concerning oxyacetylene welding, refer to Section II. After all welding has been completed, squirt hot linseed oil into the interior of the tube through the previously drilled No. 40 hole. Rotate the

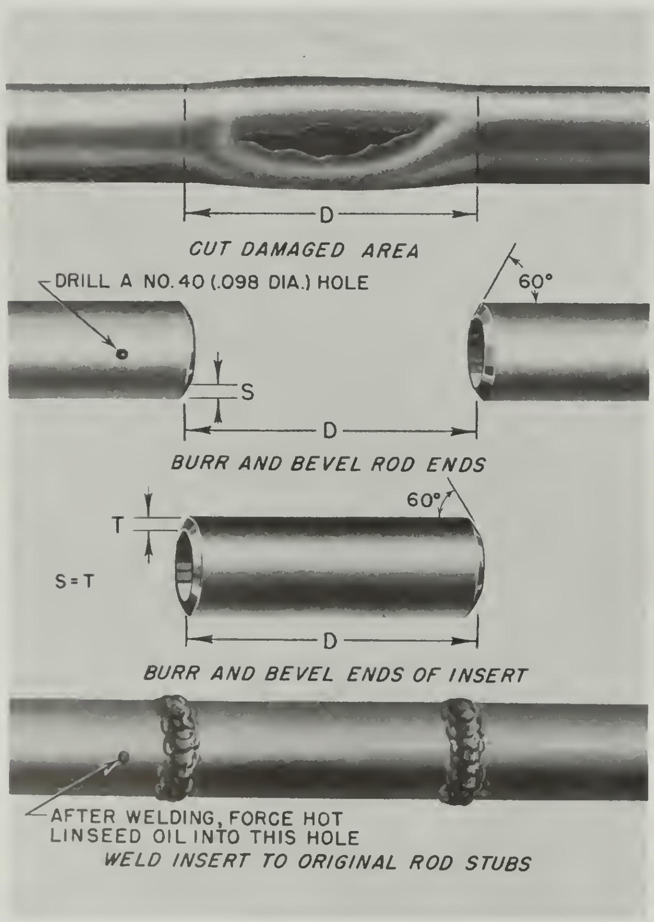


Figure 38—Control Rod Splice

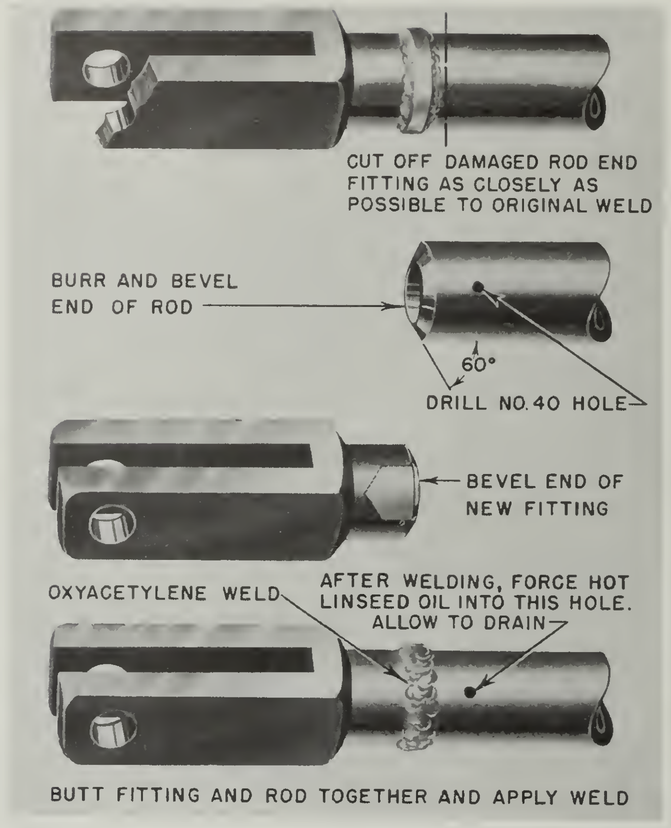


Figure 39—Control Rod Fitting Replacement

tube so that all surfaces will be covered and then allow the tube to drain. Plug the No. 40 hole with a cadmium-plated, self-tapping screw.

62. FABRICATING REPLACEMENT RODS.

Duplicate rods may be fabricated from steel tubing and the proper size and type rod end fittings. Compare those of the damaged rod with those in the illustration and determine the type needed. *(See Figure 37.)* Resistance welding should be used when this type of equipment is available. However, oxyacetylene welding will suffice. When resistance welding is used, 1/8-inch of the base metal of the tube and fitting is used at each welded joint. Therefore, when gas welding the joints, the tube must be cut proportionately shorter. When utilizing gas welding, drill a No. 40 hole in the tube and treat the tube interior for corrosion as directed in the preceding paragraph.

63. REPLACING DAMAGED ROD END FITTINGS.

Determine and obtain the proper size and type rod end fitting to replace the damaged one. With a hack saw, remove the damaged fitting by cutting the rod as close as possible to the weld securing the fitting. *(See Figure 39.)* File the rod end and the welding surface of the fitting to a 60-degree taper. Butt the fitting and rod together; and with suitable clamping devices, clamp the new fitting and the original rod in alignment. Apply a weld around the joint and treat the interior of the rod with hot linseed oil as set forth in a preceding paragraph.

ARMAMENT INSTALLATION REPAIRS

64. GENERAL.

Repairs to the bomb racks and the machine gun installations are accomplished by replacement of the worn or damaged parts. If replacement parts are not available, exact duplicates may be fabricated, but all dimensions and materials must be the same as the original part to permit proper functioning of the racks and the machine gun apparatus. *(See Figure 40.)*

Figure 40—Armament Installations

SECTION 10

REBUSHING SPECIFICATIONS

1. GENERAL.

The following list of bushings used on the AT-6 Series Airplane is divided into four tables. Table I contains all reducer bushings. Table II contains all spacer bushings. Table III contains all B Standard bushings except the B1009 bushings listed in Table II. Table IV contains all bushings which have component part numbers and are not standard bushings. In general, the ream tolerances are +.0005, -.001 unless otherwise noted, and the last dash number on all B Standard bushings denotes the length of the bushing in thirty-seconds of an inch.

TABLE I — REDUCER BUSHINGS

NOTE: AC895 reducer bushings are called out on all production drawings. However, AN912 reducer bushings may be substituted for all the following bushings.

LOCATION OF BUSHING	BUSHING PART NO.	ASSEMBLY NO.	NO. REQ.	PIPE SIZE	AN912 EQUIVALENT
INSTALLATION, SURFACE CONTROLS	AC895-80	19-52510	2	1/4 X 1/8	AN912-1
INSTALLATION, INSTRUMENTS	AC895-80	88-51001	2	1/4 X 1/8	AN912-1
ELEVATOR ASSEMBLY, COVERED	AC895-81	19-52510	2	3/8 X 1/8	AN912-3
INSTALLATION, INSTRUMENTS	AC895-82	88-51001	2	3/8 X 1/4	AN912-2
INSTALLATION, INSTRUMENTS	AC895-83	88-51001	1	1/2 X 1/4	AN912-5
INSTALLATION, INSTRUMENTS	AC895-84	88-51001	1	1/2 X 3/8	AN912-4

TABLE II — SPACER BUSHINGS

NOTE: First dash number denotes bolt size of bushing.
Second dash number denotes length of bushing in thirty-seconds of an inch.
R denotes reamed bushing.

LOCATION OF BUSHING	BUSHING PART NO.	ASSEMBLY NO.	NO. REQ.	OUTSIDE DIAMETER	INSIDE DIAMETER OR ST'D. REAM
FUSELAGE ASSEMBLY COVERED	B1009-3-4	77-31001	1	.311	NO. 10 DRILL
FUSELAGE ASSEMBLY COVERED	B1009-3-6	77-31001	1	.311	NO. 10 DRILL
FUSELAGE ASSEMBLY COVERED	B1009-3-10	77-31001	1	.311	NO. 10 DRILL
INSTALLATION, STARTING SYSTEM	B1009-R3-16	77-45001	1	.311	.1875
INSTALLATION, FIXED GUN	B1009-3-20	55-61001	1	.311	NO. 10 DRILL
FUSELAGE ASSEMBLY COVERED	B1009-3-61	77-31001	1	.311	NO. 10 DRILL
INSTALLATION, LANDING GEAR	B1009-R4-12	77-33001	2	.3735	.250
INSTALLATION, LANDING GEAR	B1009-R5-37	77-33001	2	.436	.3125
INSTALLATION, TAIL WHEEL	B1009-R6-64	77-34001	1	.498	.375

TABLE III — B STANDARD BUSHINGS

NOTE: Ream tolerances are +.0005, −.001 except where noted.

LOCATION OF BUSHING	BUSHING* PART NO.	ASSEMBLY NO.	NO. REQ.	DIMENSION TO REAM BEFORE BUSHING	DIMENSION TO REAM AFTER BUSHING
INSTALLATION, SURFACE CONTROLS	B1054-R4	55-52001	1	.375	.250
INSTALLATION, SURFACE CONTROLS	B1054-R5	55-52001	4	.375	.250
INSTALLATIONS, GENERAL	B1054-R8		13	.375	.250
INSTALLATION, HYDRAULIC SYSTEM	B1054-RB12	77-58001	1	.375	.250
INSTALLATION, HYDRAULIC SYSTEM	B1054-RB35	77-58001	1	.375	.250
STABILIZER ASSEMBLY, HORIZONTAL	B1055-6	55-21020	2	.437	.312
INSTALLATION, STARTING SYSTEM	B1055-R14	77-45001	1	.437	.312
INSTALLATION, TAIL WHEEL	B1056-38	36-34005	2	.500	.375
INSTALLATION, LANDING GEAR	B1057-R35	77-33001	2	.562	.437
FRAME ASSEMBLY, FUSELAGE FRONT	B1058-6	77-31105	4	.625	.500
INSTALLATION, TAIL WHEEL	B1058-36	36-34005	2	.625	.500
INSTALLATION, LANDING GEAR	B1060-B20	77-33001	2	.755	.625
FRAME ASSEMBLY, FUSELAGE REAR SECTION, COVERED	B1060-30	52-31169	2	.755	.625
INSTALLATION, WINGS	B1062-32	36-52612	1	1.000	.875
INSTALLATION, FURNISHINGS	B1204-6	88-53001	1	.312	.187
COVERED ASSEMBLY, WING CENTER SECTION	B1204-R20	88-13000	2	.312	.187

**EXPLANATION OF CODING:*

B1054-27 equals a steel bushing 27/32-inch long, reamed in place.
B1054-B27 equals a bronze bushing 27/32-inch long, reamed in place.
B1054-RB27 equals a bronze bushing 27/32-inch long, reamed before assembling.

TABLE IV — BUSHINGS, COMPONENT PARTS

NOTE: Ream tolerances are +.0005, −.001 except where noted.

LOCATION OF BUSHING	BUSHING PART NO.	ASSEMBLY NO.	NO. REQ.	DIMENSION TO REAM BEFORE BUSHING	DIMENSION TO REAM AFTER BUSHING
AILERON OUTBOARD HINGE BRACKET	55-16028	55-16031	1	.375	.250
CENTER AND OUTER WING FLAPS	55-18006	55-18001	10	.500 $\pm^{.002}_{.000}$	.3135 $\pm^{.002}_{.000}$
HORIZONTAL STABILIZER, FRONT FUSELAGE ATTACHING	23-21026	52-31109	2	.437	.3125 $\pm^{.001}_{.000}$
VERTICAL STABILIZER, FRONT FUSELAGE ATTACHING	55-23024	55-23022	2	.437	.3125
FRAME ASSEMBLY, FRONT FUSELAGE	19-31129-2	77-31105	1		DRILL 19/64"
FRAME ASSEMBLY, FRONT FUSELAGE	19-31129-4	77-31105	1		DRILL 19/64"
COCKPIT ENCLOSURE PANEL ROLLER	36-318115	88-31801	17		.190 $\pm^{.003}_{.000}$
LANDING GEAR SUPPORT	36-33117	77-33106	2	2.125	1.855 ±.003
LANDING GEAR RETRACTING OLEO ATTACHING FITTING	36-33119	66-33118	2	2.125	1.855 ±.003
LANDING GEAR LOCK PIN MECHANISM SUPPORT	55-33538	55-33513	1	.500	DRILL "W" (.386)
TAIL WHEEL ATTACHMENT	19-34011-2	88-34001	1		.500
CARBURETOR AIR FILTER LOCATING SHAFT	58-42092-4	58-42092	1	5/8" O.D.	DRILL 25/32" AFTER WELDING
ENGINE STARTER SHAFT BRACKET	19-45016	55-45030	1	1.484	1.281
PITOT TUBE, OUTER WING	54-51062-4	54-51062	1	5/16" O.D.	
CONTROL SURFACE LOCK HANDLE	23-52130	23-52122	1	.532 $\pm^{.005}_{.000}$	.257
CONTROL SURFACE TRIM TAB DRUM	19-52507	54-22003	2	.870	.752 $\pm^{.002}_{.000}$
FLAP EQUALIZER CONTROL CYLINDER	36-52652	36-52612	2	.437	.3125 $\pm^{.0005}_{.010}$
HYDRAULIC LANDING GEAR AND FLAP CONTROL UNIT	55-58204	55-58101	1	.437	.249 $\pm^{.000}_{.0005}$
100 LB. BOMB RACK SWAY BRACE	72-63021	84-63009	4	.3125	TAP 10-32 NF-3
100 LB. BOMB SHACKLE LEVER	84-63170	72-63141	4	.312	.1875 ±.001
RADIO ANTENNA THIMBLE ATTACHING	55-71065	88-71001	1		DRILL 1/8"

SECTION II

FINISH SPECIFICATION

1. GENERAL REQUIREMENTS.

The finish for surfaces and parts is in accordance with Specification 3-100-H, Sec. E-11, and the markings are in accordance with Specification 98-24105-P. Generally, the finish requirements may be summed up as follows:

AILERONS, EXTERIOR	ALUMINUM
AILERONS, INTERIOR AND INTERIOR PARTS	NO FINISH*
ANTIGLARE	FLAT BRONZE-GREEN NO.9
CASTINGS, FORGINGS	TO MATCH
COCKPITS	YELLOW-GREEN
COMPARTMENT, LANDING LIGHT	FLAT BLACK
COMPARTMENT, BAGGAGE	YELLOW-GREEN
COWLING, EXTERIOR & INTERIOR	ALUMINUM
ELEVATORS, EXTERIOR	ALUMINUM
ELEVATORS, INTERIOR AND INTERIOR PARTS	NO FINISH*
FAIRING	ALUMINUM
FLAPS, EXTERIOR & INTERIOR	ALUMINUM
FUSELAGE, EXTERIOR	ALUMINUM
FUSELAGE, FRAME	YELLOW-GREEN
FUSELAGE, INTERIOR PARTS, COCKPITS	YELLOW-GREEN
FUSELAGE, INTERIOR PARTS, AFT OF BAGGAGE COMPARTMENT	NO FINISH*
HORNS, CONTROL	TO MATCH
INSTRUMENT PANELS	FLAT BLACK
LANDING GEAR	ALUMINUM
PROPELLER, REAR FACE	FLAT BLACK
RUDDER, EXTERIOR BALANCE PORTION	ALUMINUM
RUDDER, INTERIOR AND INTERIOR PARTS	NO FINISH*
RUDDER, EXTERIOR REAR OF HINGE	SPEC. 98-24102-L
SEATS	ALUMINUM
STABILIZERS, EXTERIOR	ALUMINUM
STABILIZERS, INTERIOR AND INTERIOR PARTS	NO FINISH*
STEPS, PLATES	ALUMINUM
STRUTS, LANDING GEAR	ALUMINUM
TAIL WHEEL FORK	ALUMINUM
TANKS, EXTERIOR	ALUMINUM
WALKWAYS	BLACK
WHEELS	ALUMINUM
WINGS, EXTERIOR	ALUMINUM
WINGS, INTERIOR AND INTERIOR PARTS	NO FINISH*
WINGS, INSIGNIA	SPEC. 98-24102-L

*Finish for corrosion prevention shall be in accordance with the detail sections listed below. Whenever the peculiarity of a particular part or assembly prohibits the employment of the specified finish, the part or assembly shall be given a high degree of protection as is consistent with its proper functioning for its intended use.

2. FINISH AND FINISH MATERIAL SPECIFICATIONS.

The following are the U.S. Army Air Forces Specification numbers (except as noted) for the finish materials which are quoted in the following detail finish requirements:

MATERIAL	SPECIFICATION NO.
CADMIUM PLATE	AN-QQ-P-421†
CEMENT, BLACK "PLYOSYN"	26562
CLEANING, PRIOR TO FINISHING	98-20007
DOPE, CELLULOSE NITRATE, CLEAR	AN-TT-D-514
DOPE, CELLULOSE NITRATE, CLEAR (FOR ALUMINIZED DOPE)	AN-TT-D-551
ENAMEL, AIRCRAFT	3-98
FINISH, AIRCRAFT METAL PARTS	3-100‡
FINISH, AIRCRAFT WOOD SURFACES	24115
LACQUER, CELLULOSE NITRATE	AN-TT-L-51
LACQUER, CLEAR	3-158
LACQUER, FLAT BLACK	3-158
LACQUER, FLAT BRONZE-GREEN NO. 9	3-158
LACQUER, GLOSS WHITE	3-158
LACQUER, YELLOW-GREEN	3-100-H‡
MARKINGS, AIRCRAFT	98-24105
MG PROTECTIVE TREATMENT	98-20010A; E-3B
OIL, LINSEED	JJJ-O-336
PASTE, ALUMINUM	AN-TT-A-461§
PRIMER, ZINC CHROMATE	AN-TT-P-656⊕
SEALER, WOOD, LIQUID	14113
SURFACER, WOOD, LIQUID	14115
ZINC PLATE	AN-P-32†

†Zinc plating may be used in lieu of cadmium on all parts except cable terminals and steel sheet .0625 inch or thinner which is used structurally.

‡Yellow-green finish shall be formulated with a suitable tinting material. (Sherwin-Williams PX 3076 Tinting Paste or equivalent)

⊕Zinc chromate primer may be tinted with a suitable tinting material.

§The use of aluminum paste shall be restricted to pigmentation of clear dope, lacquer, or varnish for final coats on fabric, wood, surfaces of uncamouflaged airplanes, aircraft wheels, and metal only where specifically required.

11

3. FINISH CODE.

The following code of finishes, applicable to this airplane and conforming in detail to the above finish specification, facilitates the method of specifying the complete finish and color on all North American Aviation, Inc., drawings.

CODE

F – FINISH, PREFIX DESIGNATION
A – ALUMINUM AND ALUMINUM ALLOYS
C – COPPER AND COPPER ALLOYS
G – GENERAL
M – MAGNESIUM AND MAGNESIUM ALLOYS
S – STEEL
T – TEXTILE

EXAMPLE: FS-20 – Finish No. 20 for steel.

ALUMINUM AND ALUMINUM ALLOYS

4. GENERAL FA-0

Before assembly, do not anodize any structure, but prime all structures with one coat of zinc chromate primer. However, it is to be noted that alclad, 2S, 3S, and 52S shall receive no primer nor other finish except between the adjoining surfaces of riveted skin butt or skin lap joints or as required to match adjoining surfaces. After complete assembly, finish structures to match adjoining surfaces.

5. INTERIOR CLOSED MEMBERS FA-20.

Before assembly, do not anodize any structure. Do not apply primer or subsequently finish alclad, 2S, 3S, or 52S, except between riveted skin butt or skin lap joints or where it forms part of a finish surface. Apply zinc chromate primer to all other members. After complete assembly, apply two coats of yellow-green* lacquer to the cockpits and baggage compartment.

6. INTERIOR SURFACES, PARTS, AND OPEN MEMBERS FA-21.

Before assembly, do not anodize any structure. Do not apply primer or subsequently finish alclad, 2S, 3S, and 52S, except between riveted skin butt or skin lap joints or where it forms part of a finished surface. Apply zinc chromate primer to all other members. After complete assembly, apply one coat of zinc chromate primer* to the surfaces of the ailerons, elevators, rudder, and attaching parts contacting doped fabric, with the exception of alclad, 2S, 3S, and 52S. Apply two coats of yellow-green* lacquer to the cockpits, baggage compartment, attaching parts, and two coats of flat black lacquer to the landing light compartment. There are no further requirements for the bolting angle cover strip, fuselage rear of the baggage compartment, wings, and tail group. The carburetor air scoop shall not be finished on the interior and no finish or buffing is required for the seats. Buff the entire frame of 53S cockpit enclosure.

*See footnotes, Paragraph 2.

7. ENGINE COMPARTMENT SURFACES FA-23.

Before assembly, do not anodize any surface. Do not apply primer to alclad, 2S, 3S, and 52S, except between the surfaces of riveted skin butt or skin lap joints. Prime all other structures. After complete assembly, apply two coats of clear lacquer* to all surfaces with the exception of alclad, 2S, 3S, and 52S.

8. EXTERIOR SURFACES, PARTS, AND MEMBERS FA-25.

Before assembly, do not anodize any part. Alclad, 2S, 3S, and 52S will receive no primer nor further finish, except where it forms a part of a finished member or surface or between the adjoining surfaces of riveted skin butt or skin lap joints. Also, the cockpit enclosure (53S and corner castings) shall receive no primer; however, all other surfaces should receive one coat of zinc chromate primer. After complete assembly, apply two coats of flat bronze-green No. 9 antiglare. Finish entire frame of cockpit enclosure (53S and corner castings) by buffing. With the exception of alclad, 2S, 3S, and 52S, apply two coats of clear lacquer* to cowling, fairing, fuselage, landing gear, tail group, wings, and attaching parts (including flaps, flap bays, and wheel wells). Only the rear surfaces of the propeller shall receive two coats of flat black lacquer.

9. INSTRUMENT PANELS FA-28.

Before assembly, do not anodize any structure. Apply one coat of zinc chromate primer to panels. After complete assembly, apply two coats of flat black lacquer to the direct lighting portion and two coats of gloss white lacquer to the indirect lighting portion.

10. ELECTRICAL AND RADIO JUNCTION BOXES AND CONDUIT FA-29.

Before assembly, do not anodize any structure. No finish is required, except for the boxes and conduit in cockpits and baggage compartment which shall receive one coat of zinc chromate

ITEM NO.	PART NO. MOTOR SIDE	PART NO. FIREWALL SIDE	DESIGNATION	APPROXIMATE POSITION OF CONNECTION
1	77-48812	TYPE D-3 FUEL UNIT	FEED LINE FROM CARBURETOR TO RELIEF VALVE ON TOP OF HAND FUEL PUMP.	FIREWALL, LOWER L. SIDE
2	77-48810	TYPE D-3 FUEL UNIT	FUEL LINE FROM ENGINE-DRIVEN FUEL PUMP TO BOTTOM OF HAND FUEL PUMP.	FIREWALL, LOWER L. SIDE
3	77-51825	TYPE C-3 SPEC. 33215 FUEL PRESS. SIGNAL	FUEL PRESSURE LINE FROM CARBURETOR TO FIREWALL.	FIREWALL, LOWER L. SIDE
4	55-48829	55-48827	ENGINE PRIMER LINE AT FIREWALL.	FIREWALL, UPPER R. SIDE
5	77-51821	77-51822	MANIFOLD PRESSURE LINE FROM TOP OF ENGINE TO FIREWALL.	FIREWALL, UPPER R. SIDE
6	77-51829	77-51830	OIL PRESSURE LINE FROM OIL PUMP TO FIREWALL.	FIREWALL, UPPER R. SIDE
7	55-58507	55-58508	HYDRAULIC PRESSURE LINE FROM ENGINE-DRIVEN HYDRAULIC PUMP TO CONNECTORS AT LEFT LOWER TUBE OF ENGINE MOUNT.	FIREWALL, LOWER L. SIDE
8	55-58503	55-58505	HYDRAULIC RETURN LINE FROM ENGINE-DRIVEN HYDRAULIC PUMP TO CONNECTORS AT LEFT LOWER TUBE OF ENGINE MOUNT.	FIREWALL, LOWER L. SIDE
9	77-51081	77-51081	THERMOCOUPLE LINE FROM CYLINDER NO. 5 TO CONNECTOR ON FIREWALL.	FIREWALL, UPPER R. SIDE
10	77-51805	AC882-10-10	VACUUM LINE TO FIREWALL.	FIREWALL, UPPER L. COR.
11	U1050-B1	B-2 SPEC. 27943	FRONT AND REAR OIL TEMPERATURE LINES TO CAPILLARY BULBS IN "Y" VALVE.	ENGINE, LOWER R. SIDE
12	55-51845	A-7 SPEC. 27992	OUTLET EXHAUST GAS SAMPLING LINE TO EXHAUST GAS ANALYZER UNIT.	FIREWALL, R. SIDE
13	77-51844	A-7 SPEC. 27992	INLET EXHAUST GAS SAMPLING LINE TO EXHAUST GAS ANALYZER UNIT.	FIREWALL, R. SIDE
14	U1050-B1	77-47810	OIL DILUTION LINE TO "Y" VALVE.	FIREWALL, LOWER R. SIDE
15	55-43802	28-43014	THROTTLE CONTROL ROD TO BELLCRANK AT FIREWALL.	FIREWALL, LOWER L. SIDE
16	55-43804	55-43012	MIXTURE CONTROL ROD TO BELLCRANK AT FIREWALL.	FIREWALL. LOWER CENTER
17	65-43805	77-43030	CARBURETOR-AIR CONTROL ROD TO BELL-CRANK AT FIREWALL.	FIREWALL, BOTTOM SIDE
18	55-44801	55-44011	PROPELLER CONTROL ROD TO BELLCRANK AT FIREWALL.	FIREWALL, UPPER L. SIDE
19	MOTOR	94-9205-5	FRONT TACHOMETER SHAFT.	REAR OF MOTOR
20	MOTOR	94-9205-10	REAR TACHOMETER SHAFT.	REAR OF MOTOR
21	25-54080-2BB-500	88-54072	GUN SOLENOID AND GUN SIGHTS CONDUIT TO BOX ASSEMBLY.	CENTER OF FIREWALL
22	N-A-YOE1 CARBURETOR	25-54080-2AB-204	CARBURETOR MIXTURE TEMPERATURE CONDUIT TO CARBURETOR.	REAR CENTER OF MOTOR
23	25-54080-80G-330	88-54072	STARTER CONDUIT TO BOX ASSEMBLY.	CENTER OF FIREWALL
24	25-54080-6AB-394	88-54072	GENERAL CONDUIT TO BOX ASSEMBLY.	CENTER OF FIREWALL
25	25-54080-3EG-290	55-54139	IGNITION CONDUIT TO BOX ASSEMBLY.	FIREWALL, UPPER L. SIDE
26	TYPE E-4 GUN SYNCHRONIZER	77-61004	IMPULSE TUBE UNIT, TUBE TO SYN-CHRONIZER.	REAR CENTER OF ENGINE
27	55-53308	55-53338	VENTILATING SYSTEM HOT-AIR DUCT TO FIREWALL.	FIREWALL, LOWER R. SIDE
28	77-31901	77-31104	FOUR ENGINE MOUNT NACELLE ATTACHMENT POINTS.	OUTSIDE EDGE OF FIREWALL

Figure 1—Engine Disconnect Points

primer. After complete assembly, boxes and conduit in cockpits and baggage compartment shall receive two coats of yellow-green* lacquer. No other members require finish.

11. GENERAL FC-G.

Before assembly, cadmium plate* all parts except brass and bronze parts in the engine compartment, brass screws and nuts used in proximity to the compass, brass safety lockwire, and internal brass and bronze parts contacting hydraulic fluid.

12. COPPER LINES FC-20.

No finish required.

13. DISSIMILAR METALS FC-23.

Before assembly, cadmium plate* all parts and apply one coat of zinc chromate primer to all surfaces coming in contact with dissimilar metals. After complete assembly, apply two finish coats to match adjacent surfaces. Allow surfaces to dry before joining.

GENERAL FINISHES

14. CHROMIUM PLATE FG-0.

Hard chromium plate all internal steel parts contacting hydraulic fluid and communicating to exterior with the exception of corrosion-resistant steel.

15. DISSIMILAR METALS FG-5.

Before assembly, apply one coat of zinc chromate primer to all surfaces coming in contact with dissimilar metals. After complete assembly, apply two finish coats to match adjacent surfaces.

16. STEPS FG-7.

Before assembly, apply one coat of zinc chromate primer to steps. After complete assembly, finish with two coats of clear lacquer*.

*See footnotes, Paragraph 2.

17. WALKWAYS FG-8.

After assembly, apply one coat of black Plyosyn† cement and cover with 30-mesh carborundum grits. Finish with one coat of black Plyosyn cement as a sealer.

18. COIL SPRINGS FG-10.

Before assembly, do not cadmium plate the coil springs. With the exception of springs submerged in hydraulic fluid, which shall not be primed nor otherwise finished, apply one coat of zinc chromate primer to all coil springs. After complete assembly, apply two coats of yellow-green* lacquer to all springs in the cockpits and baggage compartments, and apply one coat of zinc chromate primer to all coil

*See footnotes, Paragraph 2.
†or equivalent.

Figure 2—Line Color Coding Chart

springs in the fuselage aft of the baggage compartment, wing, and tail group.

19. FLAT SPRINGS FG-11.

Before assembly, do not cadmium plate any flat springs. With the exception of springs submerged in hydraulic fluid, which shall not be primed nor otherwise finished, apply one coat zinc chromate primer to all flat springs. After complete assembly, apply two coats of yellow-green lacquer to all flat springs in the baggage compartment and cockpits.

20. ARMAMENT PROTECTION FG-13.

Ejection chutes, gun barrels, and blast tubes shall be weather-sealed by covering with a suitable cellulose acetate tape.

21. MARKINGS: LINES FG-21.

The width of each color in the marking band shall be approximately 1/2-inch. Cellulose tapes in appropriate colors coated with clear lacquer shall be used for the marking band. The following colors shall be used as the means of identification for lines. *(See Figure 2.)*

LINE	BAND
AIR SPEED	
PITOT ⊕	BLACK
STATIC ⊕	BLACK - LT. GREEN
COMPRESSED AIR	
20 P.S.I. MAX.	LT. BLUE - LT. GREEN
25 P.S.I. MIN.	YELLOW - LT. GREEN
ANTI-ICER ⊕	WHITE - RED
EXHAUST ANALYZER ⊕	LT. BLUE - BROWN
FIRE EXTINGUISHER §	BROWN
FLOTATION AND BILGE †	LT. BLUE
FUEL ‡	RED
HYDRAULIC †	LT. BLUE - YELLOW - LT. BLUE
MANIFOLD PRESSURE †	WHITE - LT. BLUE
OIL §	YELLOW
OXYGEN †	LT. GREEN
PURGING †	LT. BLUE - YELLOW
STEAM †	LT. BLUE - BLACK
VACUUM †	WHITE - LT. GREEN

22. MARKINGS: CABLES, CRANKS, AND HORNS FG-22.

Color markings for cables, cranks, and horns shall be used to identify control cables and horns or arms to which they are attached. The width of each color in the marking band shall be 1/2-inch. Cables may be banded with cellulose tape in appropriate colors and coated with clear lacquer approximately 1/2-inch behind each wrap; or if the cellulose tape is not available, the color band may be made with paint applied with a small brush. Horns and arms shall be banded with lacquer in appropriate colors approximately 1/2-inch below the fork to which cables attach.

⊕Each side of all union connections.
†Near each union.
‡Near each union and on each side of every flexible connection.
§Near each end.

CABLES, CRANKS, HORNS	BAND
AILERONS	
COCKPIT - LEFT HORN ON TORQUE TUBE	RED
COCKPIT - RIGHT HORN ON TORQUE TUBE	BLUE
WING - FORWARD HORN	RED
WING - REAR HORN	BLUE
AILERON TRIM TAB	
INBOARD CABLE (REFERRED TO COCKPIT DRUM)	BLUE
OUTBOARD CABLE (REFERRED TO COCKPIT DRUM)	RED
ELEVATORS	
CONTROL STICK SOCKET (FRONT)	BLACK
CONTROL STICK SOCKET (REAR)	YELLOW
LOWER HORN (NOSE DOWN)	BLACK
UPPER HORN (NOSE UP)	YELLOW
ELEVATOR TRIM TAB	
FORWARD CABLE (REFERRED TO COCKPIT DRUM)	YELLOW
REAR CABLE (REFERRED TO COCKPIT DRUM)	BLACK
RUDDER	
LEFT HORN (AT RUDDER)	WHITE
RIGHT HORN (AT RUDDER)	GREEN
RUDDER TRIM TAB	
FORWARD CABLE (REFERRED TO COCKPIT DRUM)	WHITE
REAR CABLE (REFERRED TO COCKPIT DRUM)	GREEN
TAIL WHEEL	
LEFT HORN (AT CABLE)	WHITE
RIGHT HORN (AT CABLE)	GREEN

23. MARKINGS: MISCELLANEOUS FG-23.

Refer to NA Dwg. 88-00010 and Spec. 98-24105. *(See Figure 3.)*

24. MARKINGS: ENGINE DISCONNECT POINTS FG-24.

The following disconnect points shall be marked by means of international-orange lacquer. The marking shall consist of a daub of paint at least 1/4-inch wide, and both units of separation are to be marked. See Figure 1 for particulars.

Note: Band at both sides of all turnbuckles.

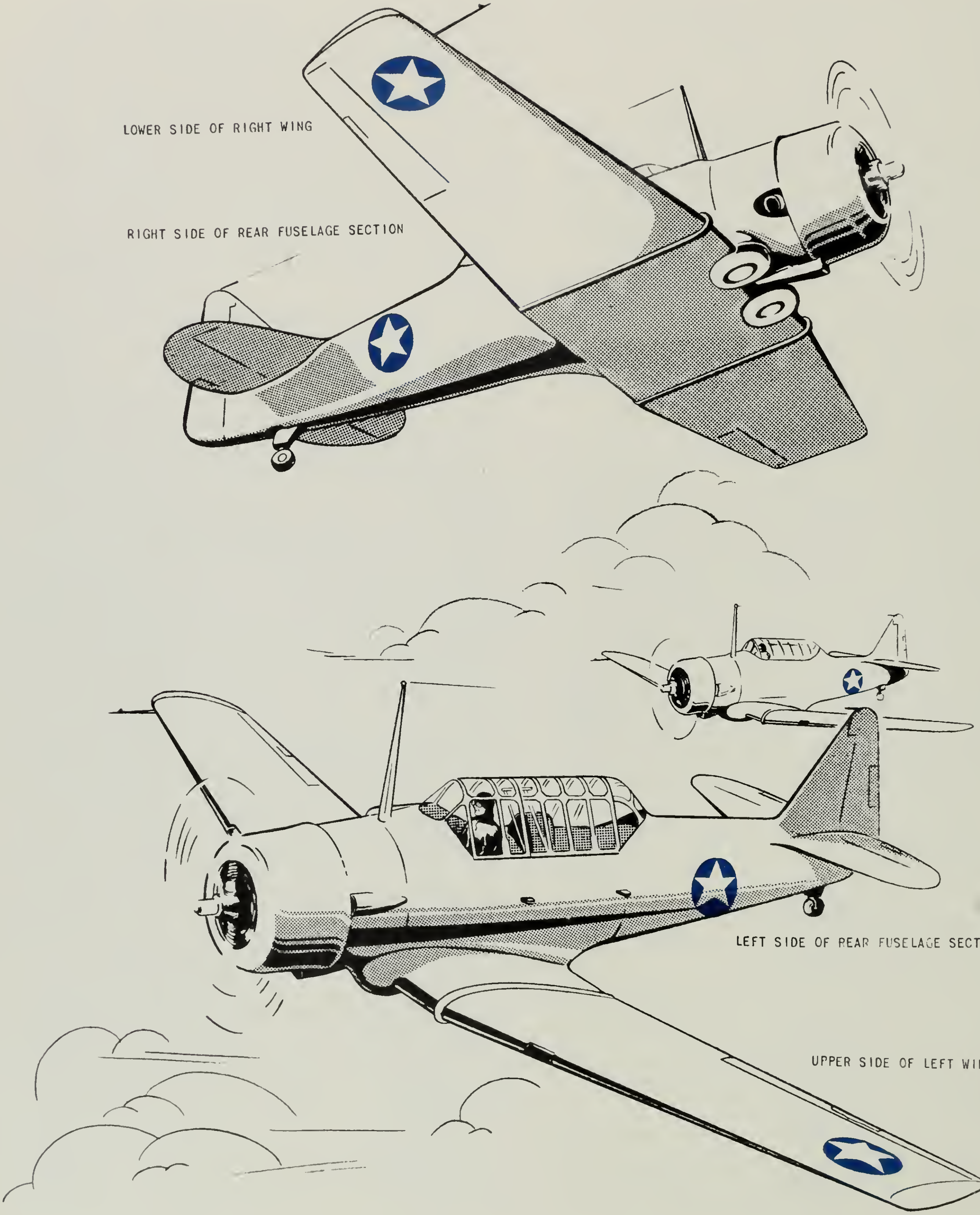


Figure 3—Airplane Insignia Locations

MAGNESIUM AND MAGNESIUM ALLOYS

25. GENERAL FM-0.

Before assembly, subject magnesium parts to protective treatment as set forth in U.S. Army Air Forces Spec. 98-20010A; E-3B. Apply one coat of zinc chromate primer to all parts and apply finish to match adjacent surfaces.

26. EXTERIOR SURFACES, PARTS, AND MEMBERS FM-20.

Before assembly, apply protective treatment to all exterior surfaces, parts, and members, as indicated in U.S. Army Air Forces Spec. 98-20010A; E-3B; and apply one coat zinc chromate primer. After complete assembly, apply an antiglare treatment of two coats of flat bronze-green lacquer No. 9. Apply two coats of clear lacquer* to the cowling, fairing, fuselage, landing gear, tail group, wings, and attaching parts (including flaps, flap bays, and wheel wells). On the SNJ-4, apply two coats of orange-yellow lacquer to the wing leading edge and upper surfaces.

27. ENGINE COMPARTMENT SURFACES, PARTS, AND OPEN MEMBERS FM-23.

Before assembly, apply protective treatment to all surfaces, parts, and open members, as indicated in U.S. Army Air Forces Spec. 98-20010A; E-3B; and apply one coat zinc chromate primer. After complete assembly, apply two coats of clear lacquer* to all surfaces.

28. INTERIOR SURFACES, PARTS, AND MEMBERS FM-25.

Before assembly, apply protective treatment to all surfaces, parts, and members, as indicated in U.S. Army Air Forces Spec. 98-20010A; E-3B; and apply one coat of zinc chromate primer. After complete assembly, finish cockpits, landing light compartment, baggage compartment, and attaching parts with two coats of flat black lacquer. To parts in view over cockpit enclosure trim, apply two coats of clear lacquer*.

PLASTICS

29. INTERIOR SURFACES, PARTS, AND MEMBERS FP-21.

With the exception of parts that are not normally visible, apply one finish coat to match adjacent surfaces.

30. EXTERIOR SURFACES, PARTS, AND MEMBERS FP-23.

Apply one coat of clear lacquer*.

31. INTERIOR OF CLOSED MEMBERS FP-24.

No finish required.

STEEL

32. GENERAL FS-0.

Before assembly, cadmium plate* all steel parts except corrosion-resistant steel. Fill closed members with hot (109°C, 165°F) raw linseed oil, drain thoroughly, and close with cadmium-plated self-tapping screws. Apply one coat of zinc chromate primer to all parts.

33. EXTERIOR SURFACES, PARTS, AND OPEN MEMBERS THAT CAN BE PLATED FS-20.

Before assembly, cadmium plate* all exterior surfaces, parts, and open members, except those of corrosion-resistant steel. Apply one coat of zinc chromate primer. After complete assembly, apply an antiglare treatment of two coats of flat bronze-green lacquer No. 9. With the exception of corrosion-resistant steel, apply two coats of clear lacquer* to the cowling, fuselage, landing gear, tail group, wings, and attaching parts (including flaps, flap bays, and wheel wells). On the SNJ-4 only, apply two coats of orange-yellow lacquer to the wing leading edge and upper surfaces.

*See footnotes, Paragraph 2.

34. EXTERIOR CLOSED MEMBERS THAT CAN BE PLATED FS-21.

Before assembly, cadmium plate* all exterior parts except corrosion-resistant steel. Apply one coat of zinc chromate primer. With the exception of corrosion-resistant steel, after complete assembly, apply two coats of

*See footnotes, Paragraph 2.

clear lacquer* to the cowling, fuselage, landing gear, tail group, wings, and attaching parts (including flaps, flap bays, and wheel wells). On the SNJ-4 only, apply two coats of orange-yellow lacquer to the wing leading edge and upper surfaces.

35. EXTERIOR CLOSED MEMBERS THAT CANNOT BE PLATED FS-22.

Before assembly, fill with hot (109°C, 165°F) raw linseed oil, drain thoroughly, and close with cadmium-plated self-tapping screws.† Apply one coat of zinc chromate primer to all parts except corrosion-resistant steel. With the exception of corrosion-resistant steel, after complete assembly, apply two coats of clear lacquer* to the cowling, fuselage, landing gear, tail group, wings, and attaching parts. On the SNJ-4 only, apply two coats of orange-yellow lacquer to the wing leading edge and upper surfaces.

36. ENGINE COMPARTMENT SURFACES FS-23.

Before assembly, cadmium plate* all steel parts except corrosion-resistant steel and the motor mounts. Closed members shall be filled with hot (109°C, 165°F) raw linseed oil, drained thoroughly, and closed with cadmium-plated self-tapping screws.† With the exception of cadmium-plated parts, apply one coat of zinc chromate primer to all steel parts. After complete assembly, apply two coats of clear lacquer* to all parts that have not been cadmium-plated with the exception of corrosion-resistant steel.

37. INTERIOR SURFACES, PARTS, AND OPEN MEMBERS THAT CAN BE PLATED FS-26.

Before assembly, cadmium plate* all steel parts except corrosion-resistant steel. Apply one coat of zinc chromate primer. After complete assembly, apply two coats of yellow-green* lacquer to the cockpits, baggage compartment, and attaching parts. The parts in view over the cockpit enclosure trim shall receive two coats of clear lacquer*. Apply two coats of flat black lacquer to the landing light compartment.

38. INTERIOR CLOSED MEMBERS THAT CANNOT BE PLATED FS-27.

Before assembly, the interior closed members shall be filled with hot (109°C, 165°F) raw linseed oil, drained thoroughly, and closed with cadmium-plated self-tapping screws.† Apply one coat zinc chromate primer to all steel parts except corrosion-resistant steel. After complete assembly, apply two coats of yellow-green* lacquer to the cockpits, baggage compartment, and attaching parts. The parts in view over the cockpit enclosure trim shall receive two coats of clear lacquer*. Apply two coats of flat black lacquer to the landing light compartment.

39. INTERIOR CLOSED MEMBERS THAT CAN BE PLATED FS-28.

Before assembly, cadmium plate* all interior closed members except corrosion-resistant steel. The interior closed members shall be filled with hot (109°C, 165°F) raw linseed oil, drained thoroughly, and closed with cadmium-plated self-tapping screws. Apply one coat of zinc chromate primer to all steel parts. After complete assembly, apply two coats of yellow-green* lacquer to the cockpits, baggage compartment, and attaching parts. The parts in view over the cockpit enclosure trim shall receive two coats of clear lacquer*. Apply two coats of flat black lacquer to the landing light compartment.

*See footnotes, Paragraph 2.
†Flash welded parts require no oiling.

TEXTILES

40. TANK PADS AND STRAPS FT-20.

Before assembly, cover with Tolex 205 DC-FP cloth (Textile Leather Corp.) or equivalent.

41. ELEVATORS AND AILERONS FT-21 RUDDER FT-23.

After complete assembly, apply two brush coats and one spray coat of clear nitrate dope followed by two spray coats of clear nitrate dope containing a minimum of six ounces of aluminum paste per gallon of spraying material. The weight of the coats shall be between 4.0 and 5.0 oz./sq.yd.

42. ANTENNA MAST FT-24.

After complete assembly, apply three coats of clear dope and then apply two coats of aluminized dope.

43. COTTON WEBBING AND FABRIC FT-25.

No treatment required for cotton webbing and fabric.

*See footnotes, Paragraph 2.
†Flash welded parts require no oiling.

WOOD

44. GENERAL REQUIREMENTS.

Finish all wood surfaces in accordance with NA Spec. SP-1707-A. Prior to gluing or finishing any wood surface, thoroughly clean the surface to remove sawdust, grease, and any other foreign matter from the wood. Wherever possible, accomplish all gluing prior to finishing, and remove all excess glue from visible interior or exterior parts before applying any finish. Whenever the peculiarity of the part is such that it is more conveniently glued subsequent to finishing, mask all areas to be glued with Scotch Tape (or suitable equivalent). No adhesive must be left on the wood when the tape is removed. Apply doped sealing tape to all areas previously taped. Before attaching metal fittings, supports, etc., protect them from direct contact with the wood by applying two coats of sealer, or one coat of sealer and one coat of filler to the wood under the fitting.

45. TYPES OF WOOD SURFACES.

Type I - Interior of Closed Members. - Surfaces normally remaining unseen, such as wing interiors, hollow spars, the underside of floorboards, etc.

Type II - Interior Open Surfaces, Parts, and Members. - Surfaces visible to occupants of the airplane, such as fuselage compartments and parts attaching.

Type III - Exterior Surfaces, Parts, and Members. - Surfaces exposed to the weather and to view from the outside.

46. DETAIL REQUIREMENTS.

Type I - Interior of Closed Members FW-24. - Apply two coats of sealer by dipping the member into the sealer or by applying the sealer with a brush or spray gun. Allow each coat to air-dry one hour. The second coat of sealer may be aluminized with 8 ounces of aluminum paste per gallon of unthinned sealer.

Type II - Interior Open Surfaces, Parts, and Members FW-21. - Apply one coat of sealer and one coat of filler. The coat of filler may be replaced by a second coat of sealer on close-grained woods such as birch. Apply one finish coat to match adjacent surfaces.

Type III - Exterior Surfaces, Parts, and Members FW-23. - Apply two coats of sealer, one coat of filler, and one coat of surfacer. Apply one finish coat to match adjacent surfaces.

47. APPLICATION OF SEALER.

After all gluing has been accomplished, apply sealer, conforming to AAF Spec. 14113, to all Type I, II, and III surfaces. Dip the surfaces into sealer when possible or spray or brush on as practical. Do not thin sealer to less than 25% nonvolatile content. Allow each coat of sealer to air-dry at least one hour. To assure complete coverage of all areas and as a guide in sanding, it is permissible to add a suitable dye to the sealer.

48. APPLICATION OF FILLER.

With a brush, apply one coat of filler (Wipe-on Formula No. 1398 or equivalent) to all Type III surfaces and wherever required on Type II surfaces. Allow to remain 2 to 5 minutes and wipe off across grain with excelsior or burlap rag. All Type II surfaces of open-grain wood such as mahogany and all exposed end-grain surfaces that are not to be taped require one coat of filler over the regular sealer coat. On all Type II and III surfaces, fill all nail holes, staple holes, and countersunk screw holes with filler. Allow the filler coat to air-dry 2 to 3 hours as required.

49. APPLICATION OF SURFACER.

With a brush or with a spray gun, apply one coat of surfacer (conforming to AAF Spec. 14115) to all Type III surfaces and allow to air-dry for 6 hours. With dry sandpaper, sand the surfacer after it has thoroughly dried. Sand so as to produce the minimum possible thickness of surfacer coat but not deep enough to touch the filler coat beneath.

50. APPLICATION OF FINISH COATS TO TYPE II SURFACES.

With a soft-bristled brush or with a spray gun, apply one finish coat to match adjacent surfaces. Use Cellulose Nitrate Lacquer (Spec. AN-TT-L-51) or Aircraft Enamel (Spec. 3-98) as required. Apply a coat of sufficient thickness to entirely cover the finishing materials.

51. APPLICATION OF FINISH COATS TO TYPE III SURFACES.

With a soft-bristled brush or with a spray gun, apply one finish coat of clear Aircraft Enamel (Spec. 3-98) pigmented with 16 ounces of aluminum paste per gallon of unthinned enamel.

52. FINISHING EXPOSED END-GRAIN SURFACES NOT SUITABLE FOR TAPING.

Apply an extra coat of sealer in addition to the regular finish to all drain holes and end-grain surfaces where taping is not specified. When finishing at drain holes and other inaccessible places, use a stiff wire covered with cloth or use an ordinary pipe cleaner.

53. FINISHING EXPOSED END-GRAIN SURFACES SUITABLE FOR TAPING.

With a hand pinking machine or with pinking shears, cut all sealing tapes required from regulation glider fabric or equivalent that has been predoped with one heavy brush coat of clear dope (Spec. AN-TT-D-514). Cut the tape to such a size as to give a minimum lap at all points of 1/2-inch onto the straight grain surface. Apply a second brush coat of clear dope to the tape and immediately apply the tape to the exposed surface. Press the tape down firmly and eliminate all air pockets beneath the tape. Allow the tape to dry for one hour and apply an additional coat of clear dope if the nap of the material is not satisfactorily filled and sealed down.

Note: Doped tape will not adhere satisfactorily to finished areas. The finish should be sanded off, sealer coated, and the tape doped on over the sealer.

54. SANDING SEALER, FILLER, AND SURFACER COATS.

With dry fine sandpaper, lightly sand the sealer and filler coats on Type II and Type III surfaces. After the surfacer coat has completely dried on all Type III surfaces, sand the coat so as to leave the minimum possible thickness of surfacer coat, but do not sand deep enough to touch or bare the filler coat beneath the surfacer.

SECTION 12

HEAT-TREATED PARTS

1. GENERAL.

The following list of heat-treated parts includes only heat-treated castings, forgings, and steel members used in the construction of the airplane. Generally, the repair of these parts is not recommended and any extent of damage necessitates complete replacement of the damaged member. However, it is to be noted that in the event of an improvised replacement, the heat treatment of the fabricated member must produce the tensile strength given on the subsequent pages in the HEAT TREAT column. As a means of estimating the tensile strength, the following table of standard hardness numbers is outlined. Emphasis is placed on the fact that the following table gives only an approximate relationship of Brinell, Rockwell, and Shore hardness values.

BRINELL		ROCKWELL	SHORE	
DIAMETER OF IMPRESSION FOR 3000 KG LOAD AND 10 MM BALL	HARDNESS NO.	C SCALE 150 KG, 120 DIAMOND CONE	SCLEROSCOPE NO.	TENSILE STRENGTH 1000 LBS./SQ. IN.
2.85	461	47	63	229
2.90	444	46	61	220
2.95	429	45	59	212
3.00	415	44	57	204
3.05	401	42	55	196
3.10	388	41	54	189
3.15	375	40	52	182
3.20	363	38	51	176
3.25	352	37	49	170
3.30	341	36	48	165
3.35	331	35	46	160
3.40	321	34	45	155
3.45	311	32	44	150
3.50	302	31	43	146
3.55	293	30	42	142
3.60	285	29	40	138
3.65	277	28	39	134
3.70	269	27	38	131
3.75	262	26	37	128
3.80	255	25	37	125
3.85	248	24	36	122
3.90	241	23	35	119

2. MATERIAL CODE.

The following code used in the heat-treated parts list simplifies the identification of materials and their physical properties.

AA-E	- ALUMINUM ALLOY EXTRUSION	24S	FEDERAL SPEC. QQ-A-354
AA-F	- ALUMINUM ALLOY FORGING	14S	FEDERAL SPEC. QQ-A-367A GR. 5
AA-F	- ALUMINUM ALLOY FORGING	17S	FEDERAL SPEC. QQ-A-367A GR. 1
AA-F	- ALUMINUM ALLOY FORGING	A51S	FEDERAL SPEC. QQ-A-367A GR. 3
AA-T	- ALUMINUM ALLOY TUBE	24S	A.C. SPEC. 10235A
CM-B	- CHROME MOLYBDENUM STEEL BAR	4140	AN-QQ-S-752
CM-F	- CHROME MOLYBDENUM STEEL FORGING		A.C. SPEC. 57-105B
CM-R	- CHROME MOLYBDENUM STEEL BAR	X4130	AN-QQ-S-684
CM-S	- CHROME MOLYBDENUM STEEL SHEET	X4130	AN-QQ-S-685
CM-T	- CHROME MOLYBDENUM STEEL TUBE	X4130	AN-WW-T-850a
MA-C	- MAGNESIUM ALLOY CASTING		AN-QQ-M-56 COND. H.T.
MC-F	- MILD CARBON STEEL FORGING		A.C. SPEC. 57-105B
MW	- MUSIC WIRE		AN-QQ-W-441
NS-R	- NICKEL-STEEL ROD	2330	AN-QQ-S-689

3. HEAT-TREATED PARTS LIST

Part No.	Title	Material Code	Heat Treat to Tensile Strength in P.S.I. or as Noted
B1226	Eye-bolt – Ball Bearing	CM-F	125-145,000
B1190	Ring – Airplane Mooring	CM-F	125-140,000
66-13013	Angle – Wing Centersection Upper Bolting	AA-E	58,000
66-13104	Angle – Wing Centersection Front Bolting	AA-E	58,000
77-13303	Angle – Wing Centersection Fuel Tank Cover Bolting	AA-E	58,000
55-14027	Angle – Outer Wing Upper Bolting	AA-E	58,000
54-14028	Angle – Outer Wing Nose Bolting	AA-E	58,000
88-14029	Angle – Outer Wing Lower Front Bolting	AA-E	58,000
54-14030	Angle – Outer Wing Lower Rear Bolting	AA-E	58,000
55-14065	Bracket – Aileron Outboard Hinge	AA-F (17ST)	55,000
55-14066	Bracket – Aileron Inboard Hinge	AA-F (14ST)	65,000
55-14069	Bracket – Aileron Center Hinge	AA-F (14ST)	65,000
55-14244	Support – Aileron Bellcrank Bearing	AA-F (17ST)	55,000
55-16080	Bracket – Aileron Inboard Hinge	AA-F (14ST)	65,000
55-16081	Fitting – Aileron Center Hinge	AA-F (17ST)	55,000
55-18022	Hinge Fitting – Flap	AA-F (A51ST)	43,000
55-18024	Eye-bolt – Wing Flap	CM-B	125,000
52-21018	Fitting – Elevator Center Hinge Support	AA-F	65,000
52-21019	Fitting – Elevator Center Hinge Support	AA-F	65,000
55-21020	Fitting – Stabilizer Attaching	AA-F (A51ST)	43,000
52-21026	Bracket – Elevator Inboard Hinge Support	AA-F	65,000
55-22037	Hinge Fitting – Trim Tab	AA-F (17ST)	55,000
52-23012	Fitting – Rudder Upper Hinge Support	AA-F	65,000
52-23013	Fitting – Rudder Center Hinge Support	AA-F	65,000
52-23014	Fitting – Rudder Lower Hinge Support	AA-F	65,000
28-31053	Fastener – Engine Ring Cowling Rear	AA-S	62,000
66-31058-2	Catch – Fire Extinguisher Door Release	AM-S	125,000
66-310102	Hook – Engine Ring Cowling	AA-E	62,000
57-31103-2	Tube – Fuselage Station No. 1 Lower Cross	CM-T	Normalize
19-31105	Fitting – Fuselage Connection	CM-F	Normalize
55-31174	Fitting – Fuselage Connection	CM-B	125-140,000
55-31220	Step – Cockpit	MA-C	32,000
19-31231	Bracket – Fuselage Pilot's Seat Support	CM-F	Normalize
55-31383	Guide – Fuselage Front Step Tube	CM-B	125,000
55-31385	Spring – Fuselage Front Step Tube Keeper	MW	*(10 Minutes)
49-31894	Finger – Enclosure Release Handle	CM-S	160-180,000
25-31902	Fitting – Engine Mount Attaching	CM-F	Normalize
66-33109	Pin – L.G. Retract Arm Attach.	CM-B	145-160,000
36-33112	Pin – L.G. Pivot	CM-T	160-180,000
36-33117	Bushing – L.G. Support	CM-B	160-180,000
36-33119	Bushing – L.G. Attach. Fitting	CM-B	125-140,000
36-33120	Nut – L.G. Pivot Pin	CM-B	125,000
19-33447	Wrench – Special	CM-S	180-200,000
55-33467	Spring – Brake Cylinder Ext.	MW	*(20 Minutes)
36-33515	Latch – L.G. Lock Cam	CM-S	125,000
55-33525	Pin – Landing Gear Lock	CM-B	160-180,000
44-33534-2	Fitting – L.G. Lock and Hoist	CM-F	160-180,000

* – Heat treat spring at 500-550°F for period noted.

3. HEAT-TREATED PARTS LIST (CONTD.)

Part No.	Title	Material Code	Heat Treat to Tensile Strength in P.S.I. or as Noted
44-33534-3	Fitting - L.G. Lock and Hoist	CM-F	160-180,000
66-33536	Bolt - L.G. Lock and Hoist Fitting Attach.	NS-R	160-180,000
36-33541	Rod - L.G. Lockpin Push	CM-F	Normalize
36-33542	Bolt - L.G. Mechanism	CM-B	125,000
55-33544	Bellcrank Assembly - L.G. Locking Mechanism	CM-F	125,000
66-42012	Shaft - Carburetor Mixing Chamber	CM-T	125,000
51-47032	Stud - Oil Tank Sump	NS-R	125,000
49-51047	Support - Oil Separator Outer	AA-S	62,000
55-52119	Spring - Surface Lock	MW	*(20 Minutes)
55-52302	Bellcrank - Aileron	AA-F (17ST)	55,000
55-52316	Clevis - Aileron Control Rod	CM-F	125,000
55-52428-2	Rod - Rudder Brake Pedal	CM-R	125,000
19-52440	Spring - Tension Coil	MW	*(20 Minutes)
49-52441	Spring - Rear Cockpit Rudder Pedal Adjust.	MW	*(20 Minutes)
55-52610	Arm - Flap Indicator	MA-C	32,000
55-52649-2	Rod - Flap Connecting	CM-B	125,000
55-52653	Plunger - Flap Control Handle Lock Spring	NS-R	Carburize 1/32-Inch
58-53004-3	Tube - Pilot's Cockpit Seat Support	CM-T	160-180,000
49-53060	Clamp - Alemite Grease Gun	SS-S	Spring Temper
55-53061	Pin - Observer's Seat Latch	CM-B	160-180,000
49-53073	Plunger - Seat Adjusting	CM-R	125-140,000
55-53075	Tube - Observer's Seat Adjusting	CM-T	160-180,000
55-53076	Brace Assembly - Observer's Seat	CM-T	150-170,000
55-53096	Spring - Observer's Seat Latch	MW	*(10 Minutes)
55-53104	Axle - Observer's Seat	CM-B	160,000
36-54029	Bearing - L.G. Position Indicator Switch Box	CM-B	125,000
36-54033	Guide - L.G. Position Indicator Switch Box	CM-B	125,000
55-54083	Bracket - Ignition Switch Hand Support	MA-C	32,000
55-54125	Spring - L.G. Position Indicator Switch	MW	*(10 Minutes)
36-55009	Fitting Assembly - Wing Center Section Hoist	CM-F	125-140,000
36-58008-2	Fitting - L.G. Hydraulic Strut	CM-B	125,000
88-58101	Body - Hydraulic L.G. & Flap Control Unit	AA-F (17ST)	55,000
55-58137	Spring - Hydraulic One-Way Valve	MW	*(5 Minutes)
55-58143	Spring - Hydraulic L.G. & Flap Relief Outer	MW	*(20 Minutes)
55-58144	Spring - Hydraulic L.G. & Flap Relief Inner	MW	*(10 Minutes)
55-58152	Spring - Control Unit One-Way Valve	MW	*(20 Minutes)
55-58196	Spring - Pressure Control Return	MW	*(20 Minutes)
55-61040	Screw - Ammunition Box & Ejection Chute Adjust.	NS-R	125,000
59-73006	Bow - Instrument Flying Hood	AA-T	62,000
59-73007	Bow - Instrument Flying Hood	AA-T	62,000

* - Heat treat spring at 500-550°F for period noted.

ALPHABETICAL INDEX

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