SECTION 16

# STRUCTURAL REPAIR

Table of Contents



## GENERAL.

### Type of Construction.

The 310 is an all-metal aircraft of semimonocoque type construction with the skin carrying a portion of all structural loads. The fuselage is comprised of a forward cabin section and a tailcone. It is constructed of formed bulkhead rings, stringers, and stiffeners all of which are riveted to the external skin. The wing, horizontal stabilizer and vertical fin are built up around two main spars, with ribs, formers and riveted skin forming the basic structure. Torsional stiffness of this structure is afforded by the skin closure of areas between the spars forming enclosed "boxes." Each movable surface consists of a hinge support spar with ribs, formers and riveted outer covering skin.

Ground Handling.

Leveling, jacking and other ground handling details are covered in Section 2.

Investigation of Damage.

After a thorough cleaning of the damaged area, all structural parts should be carefully examined to determine the extent of damage. Frequently the force causing the initial damage is transmitted from one member to the next, causing strains and distortions. Abnormal stresses incurred by shock or impact forces on a rib, bulkhead or similar structure may be transmitted to the extremity of the structural member, resulting in secondary damage such as sheared or stretched rivets, elongated bolt holes, canned skin plate or bulkheads. Points of attachment should be examined particularly for distortion and security of fastenings in the primary and secondary damaged areas.

Definition of Damage.

Structural Damage to the aircraft is divided into the following classifications:
a. Negligible Damage shall be considered damage that will not affect the airworthiness of the aircraft and can be permitted to exist as is or can be corrected with a simple repair such as removing dents, burnishing scratches and stop drilling cracks in nonstructural parts.
b. Damage Repairable by Patching will be considered damage that may be repaired by covering or reinforcing a portion of the aircraft.
c. Damage Repairable by Insertion will be considered damage requiring replacement of a section with the correct repair material.
d. Damage Necessitating Replacement of Parts will be considered as damage not repairable by patching or insertion, but that may be repaired by installing a new or reconditioned part. If a part or area of an assembly is damaged to the extent that it requires replacement, and a replacement cannot be made because of tooling or jig requirements, the entire assembly must be replaced.

Preparing Damaged Area for Repairs.

To prepare an area for repair, examine and classify the damage. Make a thorough check before beginning repairs. In some cases a damaged part may be classified as needing replacement when after removal, closer inspection indicates the part may be repaired. Take more time for the damage estimate and save man-hours on repairs. To prepare a damaged area for patch or inserting repairs:
a. Remove all ragged edges, dents, tears, cracks, punctures and similar damages.
b. Leave edges, after removal of damaged area, parallel to any square or rectangular edges of the unit..
c. Round all square corners.
d. Smooth out abrasions and dents.
e. Brush Iridite all rough edges and scratches with a solution of Iridite mixed in a ratio of 1 ounce of Iridite to 1 gallon of water and rinse thoroughly.
f. Apply two coats of zinc-chromate primer to all internal surfaces and edges lapping over another.

NOTE

Damage adjacent to a previous repair requires removal of the old repair and inclusion of the entire area in the new repair.

Control Surface Rebalancing Data.

The control surfaces of the aircraft have been 100% statically balanced. After each repair or painting of the control surfaces they must be rebalanced. Correct balance is restored by the addition or removal of lead ballast weights in the counterbalance sections of the surfaces.

WING.

The wings are all-metal, full cantilever, semimonocoque type construction, utilizing two main spars. Each wing consists of a wing panel, aileron, flaps, engine nacelle, wing tip fuel tank, and main landing gear. The landing gear is attached to and retracts into the wing.

Access Openings.

Access openings with removable cover plates are located in the underside of the wing between the root rib and the tip section. These openings afford access to the aileron bellcranks, flap bellcranks, electrical wiring, pulleys, cables and inspection of internal structure. When work is done on the trailing edge wing structure in the flap area, partial access can be provided by lowering the flaps. Outboard of this area, the trailing edge wing structure can be made available for repair by removing the aileron.

Wing and Horizontal Stabilizer Angle of Incidence.

Angle of incidence is defined as the angle between the wing or stabilizer chord line and aircraft waterline (aircraft level longitudinally). Stabilizers do not have twist. Wings have a constant rate of twist from the wing root rib to the tip rib. All twist is between these two ribs. The amount of twist between these points is the difference between the angle of incidence at the root rib and the angle of incidence at the tip. Refer to Section 3 to check wing twist.

Wing Skin.

All wing, aileron, and flap skin thickness and temper are listed in figure 16-10.

Negligible Damage.

Any smooth dents in the wing skin that are free from cracks, abrasions and sharp corners, which are not stress wrinkles and do not interfere with any internal structure or mechanism may be considered as negligible damage. In areas of low stress intensity, cracks, deep scratches, or deep sharp dents, which after trimming or stop drilling can be enclosed by a two-inch circle, can be considered as negligible if the damaged area is at least one diameter of the enclosing circle away from all existing rivet lines. Stop drilling is considered a temporary repair.

Repairable Damage.

Skin damage ahead of the front spar and also where the optimum in appearance is desired should be repaired by the insertion method. Typical insertion repairs are illustrated in figure 16-16. Skin damage aft of the front spar which exceeds the negligible damage limit but is not extensive enough to necessitate replacement of a skin panel can be repaired by patching. Typical wing repairs are illustrated in the back of the Section.

Damage Necessitating Replacement of Parts.

In case the skin is extensively damaged, repairs should be made by replacing an entire sheet panel from one structural member to the next. The repair seams should be made to lie along stiffening members, or bulkheads, and each seam should be made exactly the same in regard to rivet size, spacing and rivet pattern as the manufactured seam at the edges of the original sheet. If the two manufactured seams are different, the stronger one should be copied.

Wing Ribs.

All ribs except those exposed to the wheel well contain flanged lightening holes. Flanged upper and lower edges of all ribs serve as cap-strips in addition to providing rigidity to the rib. The skin riveted directly to each rib flange provides the cellular strength for each successive rib bay. The nose, center and trailing edge rib segments are riveted together through the front and rear spars to form the basic airfoil sections. Spanwise alclad stringers stiffen the skin between ribs.

Negligible Damage.

Refer to negligible damage paragraph under wing skin.

Repairable Damage.

Repairs for wing rib webs and flanges are shown in figures 16-25 and 16-26. Before repairing is attempted, all cracks or deep scratches must be stop-drilled with a 3/32-inch drill, and all sharp corners and ragged edges must be trimmed and deburred.

Damage Necessitating Replacement of Parts.

Parts such as stiffeners, small ribs, clips and brackets should be replaced if their damage exceeds that specified as negligible. These parts, due to their size, are usually impractical to repair. In many instances, the time required to replace the damaged part may be considerably less than the time required to repair it. This should be considered carefully prior to making any repair whether it be only a small part or a complete component.

Wing Spars.

Repair of spar damage affecting the alignment of the wing spar should not be attempted in the field. Permissible spar repairs are illustrated in the back of this Section.

Flaps and Ailerons.

Negligible Damage.

Minor skin dents and nicks are considered negligible and can be worked out by burnishing.

Repairable Damage.

Skin damage exceeding that considered negligible damage can be repaired by patching. Typical skin repairs are illustrated in the back of the section.

Damage Necessitating Replacement of Parts.

Warped and cracked skin, ribs, hinge brackets and torque tubes are replaceable items. Any damage that covers more than half of the unit will require replacement of the entire unit.

TAIL GROUP.

The all-metal tail group is of full cantilever design, consisting of the conventional arrangement of vertical fin and rudder, horizontal stabilizer and elevators. The right elevator and the rudder both contain flight adjustable trim tabs, actuated by a system of cables and pulleys controlled from the pilot's tab control wheels.

Vertical Fin and Dorsal Group.

The vertical fin and dorsal area are constructed jointly to form a single unit. Basically the unit consists of formed sheet metal spars and ribs to which the outer skin is attached. The front spar is reinforced at its root end and drilled to facilitate the installation of two attachment bolts. Stiffness to the entire fin and dorsal assembly is provided by the attachment of the skins and the forward leading edge skin.

Negligible Damage.

Refer to negligible damage paragraph under wing skin.

Repairable Damage.

Repair of the skins, ribs and spars can be accomplished as illustrated in the back of the section. Access to the internal fin structure is best gained by removing the skin attaching rivets on one side of the rear spar and springing back the skin.

Damage Necessitating Replacement of Parts.

Extrusion, hinge brackets and small ribs should be replaced rather than repaired. In general, where parts are available, the easiest and most satisfactory repairs can be accomplished by replacing the damaged parts.

Rudder.

Negligible Damage.

Minor skin dents and nicks are considered negligible and will be worked out by burnishing.

Repairable Damage.

Skin damage, exceeding that considered negligible damage, can be repaired by patching. Typical skin repairs are illustrated in the back of the section.

Damage Necessitating Replacement of Parts.

Warped and cracked skin, ribs, hinge brackets and torque tubes are replaceable items. Any damage that covers more than half of the rudder will require replacement of the rudder.

Horizontal Stabilizer.

Negligible Damage.

Refer to negligible damage paragraph under wing skin.

Repairable Damage.

Skin damage, exceeding that considered negligible damage, can be repaired by patching. Typical skin repairs are illustrated in the back of the section. Repairs to spars should consist of channels formed of the same material and bend radius as the spar and extending at least three inches each side of the damaged area. Access to the internal stabilizer structure may be gained by removing a portion of the rivets along the rear spar and ribs and springing back the skin. By using the proper bucking bars through holes in the spar web, the skins may be closed with a minimum of blind rivets.

Damage Necessitating Replacement of Parts.

Extrusions, hinge brackets, stabilizer tab, spar and ribs should be replaced rather than repaired. In general, where parts are available, the easiest and most satisfactory repairs can be accomplished by replacing the damaged parts.

Elevators.

Negligible Damage.

For a description of negligible damage, refer to negligible damage paragraph under wing skin. The exception to negligible damage on the elevator surfaces is the front spar, a crack appearing in the web at the hinge fittings or in the tip rib which supports the overhanging balance weight is not considered negligible. Cracks in the over-hanging tip rib, in the area at the front spar intersection with the web of the rib, also cannot be considered negligible.

Repairable Damage.

Skin damage, exceeding that considered negligible damage, can be repaired by patching. Typical skin repairs are illustrated in the back of the Section.

Damage Necessitating Replacement of Parts.

Warped and cracked skin, ribs, hinge brackets and torque tubes are replaceable items. Any damage that covers more than half of the elevator will require replacement of the elevators.

FUSELAGE.

The fuselage is of semimonocoque construction consisting of formed bulkhead, longitudinal stringers, reinforcing channels and skin platings. The fuselage forward section consists of all the fuselage structure from the nose to station 109.375. Formed bulkheads, channels and extrusions constitute the frame members of the cabin area.

Negligible Damage.

Refer to negligible damage paragraph under wing skin.

Repairable Damage.

Mild wrinkles occurring in the upper or lower skin panels in the bay forward of the horizontal stabilizer and which extend through the corners (shoulder areas) may be repaired by the addition of a stringer. A wrinkle, which is hand removable, should be reinforced by a 1/2 x 1/2 x .050-inch 2024-T42 extruded angle. The angle should be inserted fore and aft across the center of the wrinkle and should extend to within 1/16 to 1/8-inch of the fuselage bulkheads comprising the ends of the bay. If wrinkles cannot be removed by hand, the damaged area should be repaired.

Damage Necessitating Replacement of Parts.

All forgings and castings of any material and structural parts made of steel must be replaced if damaged. Structural members of a complicated nature that have been distorted or wrenched should be replaced. Major skin damage should be repaired by replacing the entire damaged sheet.

NOTE

When replacing entire skin panels, duplication of the forward edge of the original sheet is required. In effect, this flange is a structural member, carrying specific loads across the open areas.

Bulkheads.

Bulkheads are composed of formed "U" channel sections. The principle material of construction is 2024-O alclad aluminum alloy, which after forming is heat treated to a 2024-T42 condition. All bulkheads in the fuselage are of the formed sheet metal or the reinforced formed sheet metal type.

Cracked Bulkhead Webs or Flanges.

Acceptable methods of repairing various types of cracks occurring in service are shown in the back of the section. Small holes (3/32-inch) should be drilled at the extreme ends of the cracks to prevent further spreading. Reinforcement should be added to carry the stresses across the damaged portion and stiffen the joints. The condition causing such cracks to develop at a particular point may be stress concentration at that point, in conjunction with repetition of stress (such as producedby vibration of the structure). The stress concentration may be due to defects such as nicks, scratches, tool marks and initial stresses or cracks from forming or heat treating operations.

NOTE

An increase in sheet thickness alone is usually beneficial but does not necessarily remedy the conditions leading to cracking. Patch type repairs are generally employed and are usually satisfactory in restoring the original material strength characteristics.

Severely Bent, Kinked or Torn Channels.

If practical, severely bent, kinked or torn portions of bulkheads should be removed and a replacement section installed and joined at the original splice joint. If this is not justified, cutting away the damaged portion and inserting a trimmed portion of the original section, adequately reinforced by splice plates or doublers, will prove satisfactory. This is known as an insertion type repair.

LANDING GEAR.

The main gears are carried by the wings and are housed within the wing wheel wells when retracted. The nose gear retracts into the fuselage nose wheel well. Doors covering the wells are regarded as parts of the landing gear assemblies but the wells are structural features of the wings and fuselage.

Repairs of Landing Gear.

The landing gear assemblies are composed of parts that are not regarded as repairable. Minor repairs are permissible on the doors but when they are reinstalled there must be no distortion that will prevent perfect operation.

ENGINE NACELLE.

The engine nacelle group, located in each wing, is composed of the semicantilever bed-type mount, the stainless steel firewall, and the cowling group. The engine mount structure is made of .063 inch 2024-T4 aluminum and 4130 cad-plated steel. The cowling is made up of three sections; the upper, the lower which includes the left and right access doors, and the two piece nose section which fastens in the center of the cowling. All sections fasten in place with Camloc quick fasteners. The nacelle firewall is made up of stainless steel sheet with a clad aluminum angle riveted around its contour.

a. Negligible Damage.

Minor dents in the firewall and cowling, if straightened, may be classified as negligible damage. Scratches and dents should be burnished out.

Engine Firewall.

a. Repair of Stainless Steel Firewall.

The firewall may be repaired by using the clear of structure-type patch, as shown in figure 16-24A of this section, providing the patch is of the same thickness as the firewall and monel rivets are used. Maximum diameter on holes that can be patched is 2 inches. The holes should be routed and repaired in accordance with figure 16-24A. Cracks should be stop drilled and repaired in accordance with figure 16-24A. Parts having cracks extending to the edge of the part must be replaced. Maximum allowable length of cracks to be repaired is 3 inches.

Engine Firewall Sealing.

The engine firewall should be sealed with pro-seal #700 (Coast Pro-Seal Company) using the following procedures:

a. Clean area on surface to be sealed with solvent
b. Mix 1 part of Pro-seal #700 curing agent with 100 parts of Pro-seal #700 sealant.

NOTE

Sealant should be mixed by weight. It is important that accelerator be completely and uniformly dispersed throughout the base compound.

c. Using a spatula, caulking gun, or flow gun, apply a fillet of sealer along cracks, seams, joints and rows of rivets.

NOTE

If the sealing is done before the parts are mated, use enough sealing compound to completely fill the joint and wipe away the excess after parts are mated.

NOTE

If the sealant is applied with a brush or a brush flow gun, more than one coat of sealant will be necessary on very porous material. The sealant should be allowed to air dry 10 minutes between coats.

d. Pro-seal #700 is the only sealant authorized for the stainless steel firewall. If sealant other than pro-seal #700 has been used it should be removed from the firewall and resealed using only the recommended sealant.

Engine Mount.

Replacement of mount is the only recommended procedure.

Repair of Engine Cowling.

Skin, if damaged extensively, should be replaced with a section of original manufacture. Small damaged areas should be reinforced with a doubler installed on the inner side. Material selected should be of the same thickness and characteristics as the original part.

Repair of Cowling Reinforcement.

Cowling reinforcements, if damaged, should be replaced. Due to their small size and complex angles they are easier to replace than to repair.

Repair Procedures for Bonded Honeycomb Engine Cowls. (See figure 16-33.)

a. The 310 engine cowling is fabricated using a sandwich type honeycomb construction. Sometime during the life of the aircraft damage may occur to the skin and/or honeycomb which will require repairs. In the event damage does occur, the following repair procedures have been developed with the objective of equaling, as nearly as possible, the strength of the original part with a minimum loss of aerodynamic characteristics, electrical properties and minimize increase in overall weight.
b. Damages to sandwich honeycomb construction are divided into classes according to severity and possible effect upon the airframe structure. Damage classifications are as follows:

Class 1. Dents, scars, scratches, cracks, etc., in the facings not accomplished by a puncture or a fracture.

Class 2. Punctures or fractures on one facing only, possibly accompanied by damage to the honeycomb core but without damage to the opposite facing.

Class 3. Holes or damage extending completely through the sandwich, affecting both facings and the core.

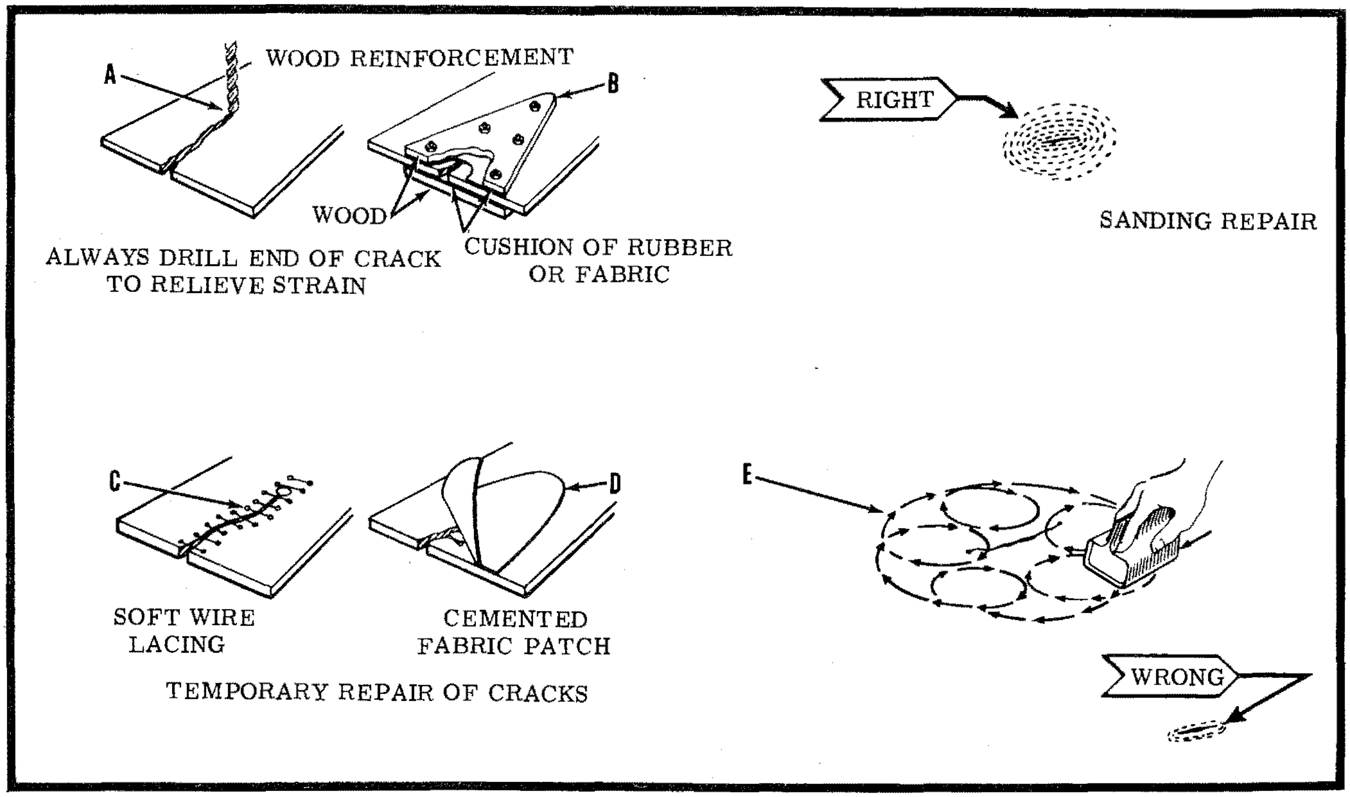



Figure 16-1. Plastic Repairs

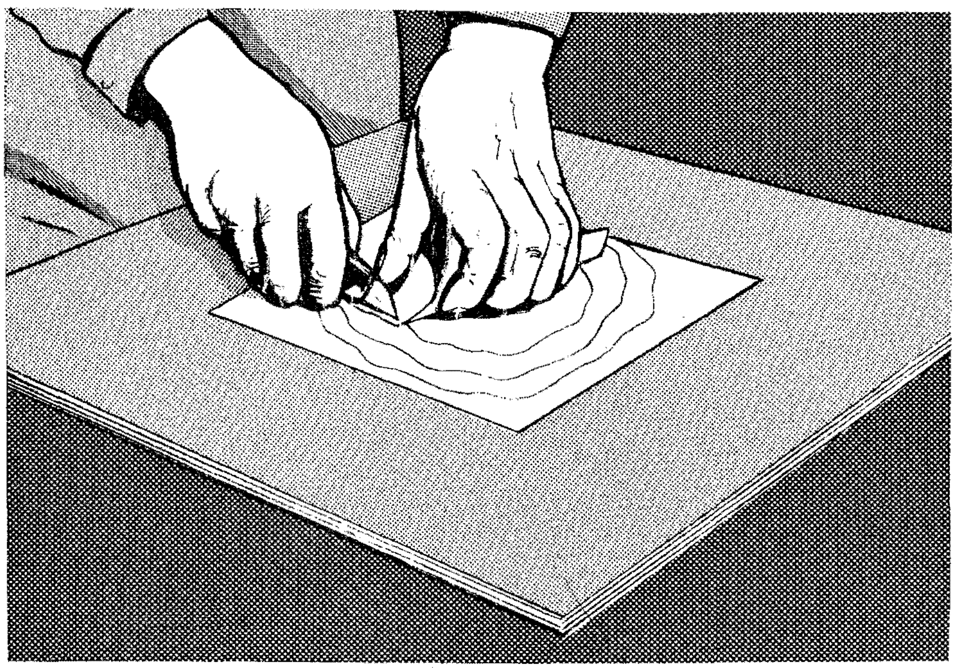


Figure 16-2. Removing Damaged Face Plies - Step Joint Method

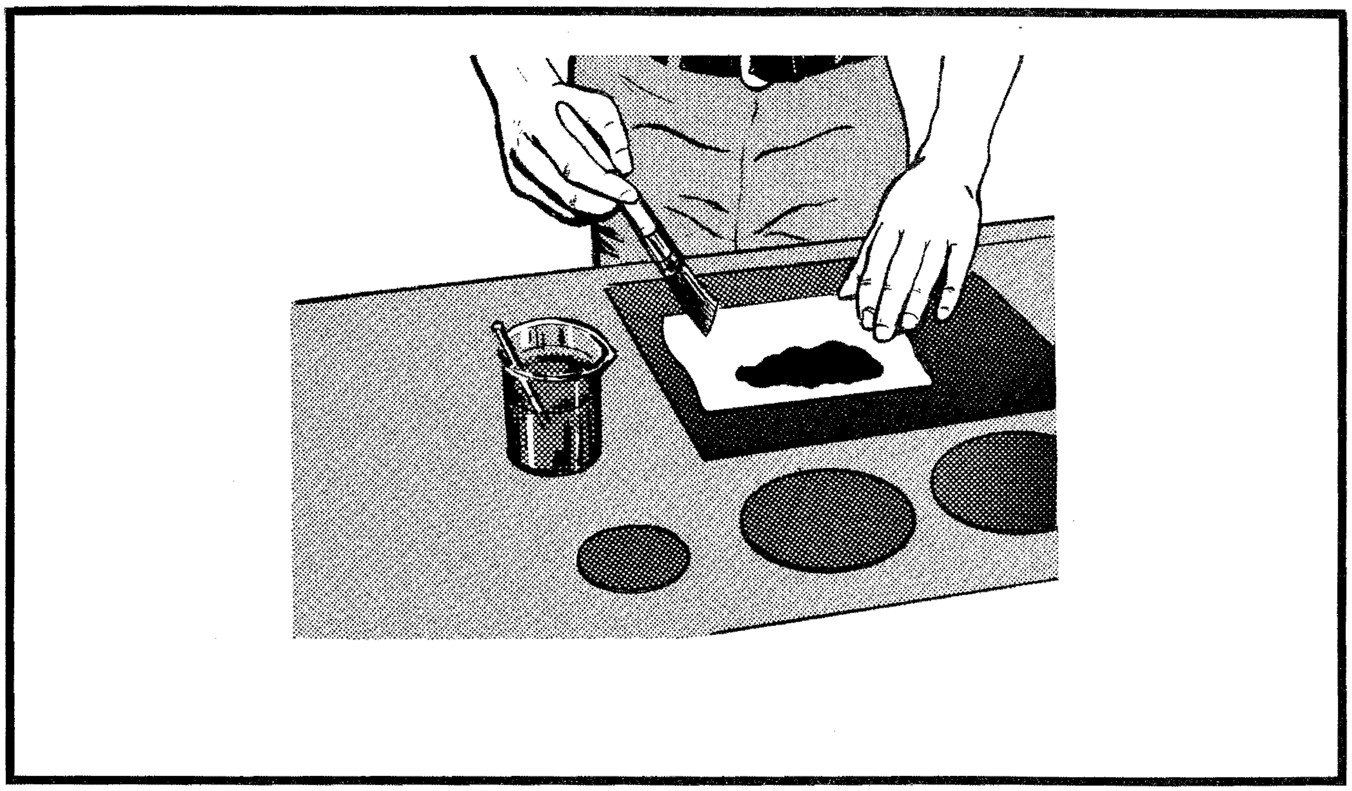


Figure 16-3. Suggested Method of Resin Impregnating Replacement Plies

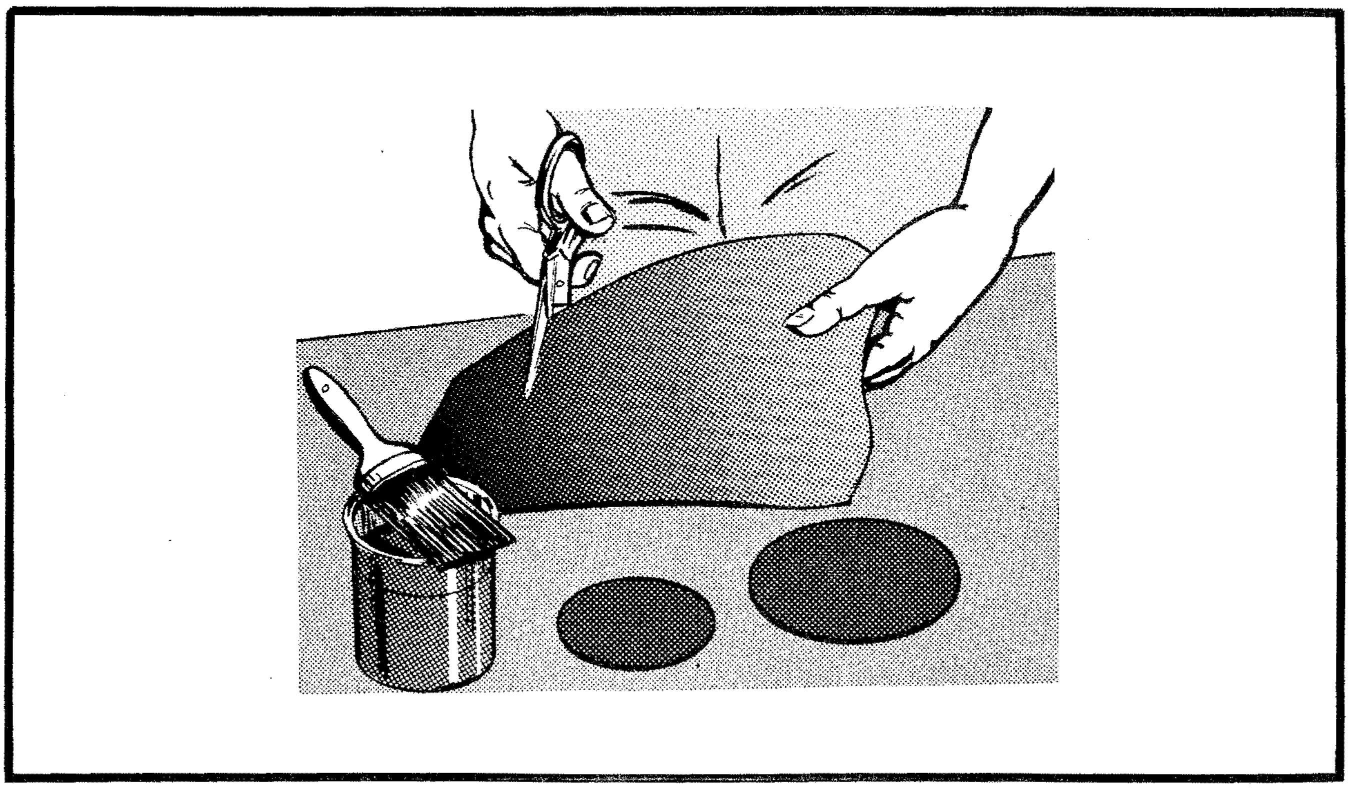


Figure 16-4. Cutting Replacement Plies from Impregnated Glass Cloth
Sandwiched Between Sheet of Cellophane

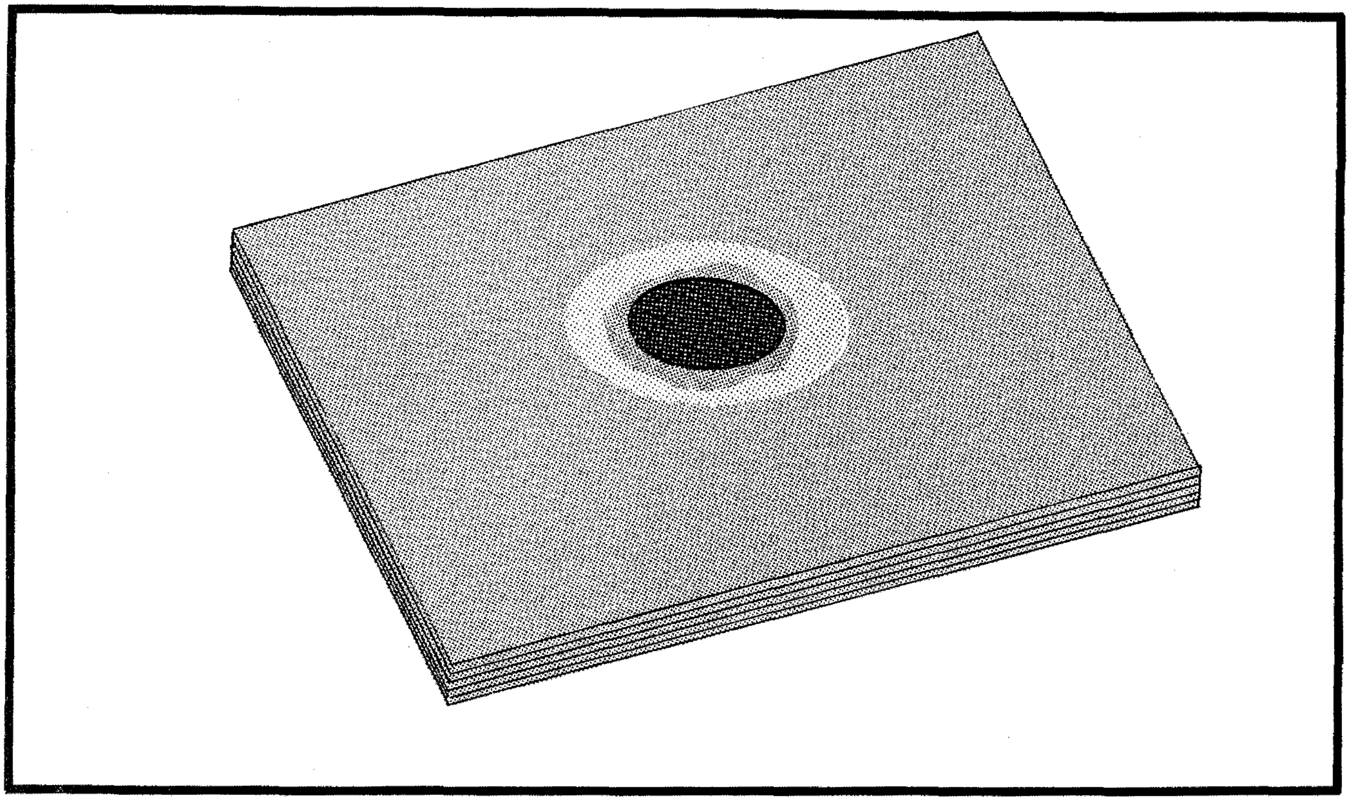


Figure 16-5. Damaged Face Plies Removed - Scarf Method

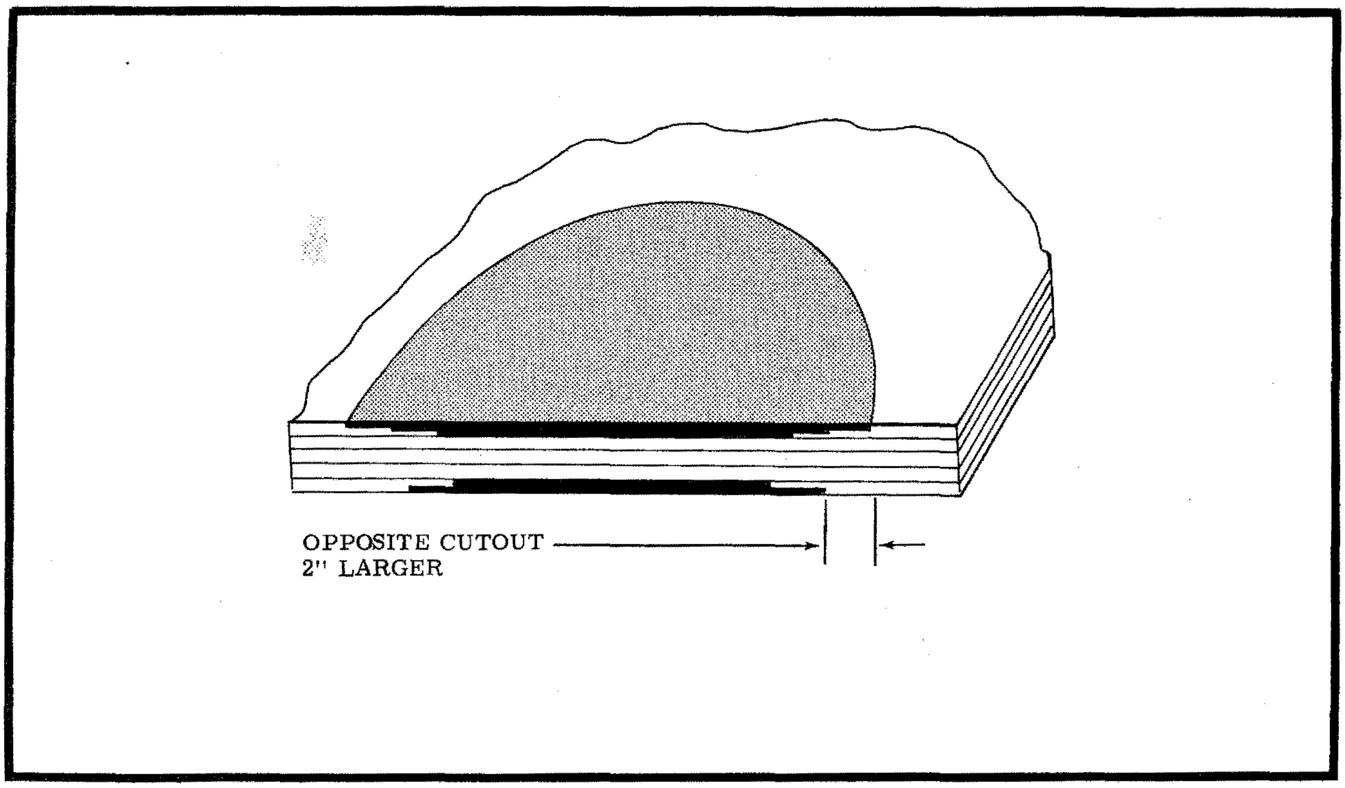



Figure 16-6. Completed Double Face Patch Repair

Class 4. Adhesive voids between skin and honeycomb core.

WARNING

Solvents used must be stored in, transported in, and used from safety containers. Adequate ventilation must be provided in storage and usage areas. The solvents specified are flammable and caution to prevent fires must be taken. No smoking, sparks or open flames shall be permitted in the immediate area where the solvents are being used. Storage and usage areas shall be free from excessive heat, sparks, and open flames when possible and practical, rubber gloves shall be worn when performing solvent operations.

c. Repair of Class 1 damage shall be as follows:

1. Class 1 damage resulting in a hole not exceeding 1-1/2 inches diameter, dents, scratches or scars of .030 inch deep shall be repaired as follows:

(a) Remove paint to bare metal using either 400 or 600 grit wet sandpaper.

(b) Mask area adjacent to sanded area.

(c) Mix thoroughly equal portions, by volume, of Epon 828 and Versamid 125. Add aluminum powder until a thick non-flowing paste is obtained.

NOTE

Prepare only that quantity of material that will be used in 30 minutes.

(d) Fill damaged area with mix and smooth with a putty knife or spatula.

(e) Allow the mix to cure at room temperature until hard (approximately 4 hours).

(f) Wet sand the repaired area with 400 grit wet sandpaper until smooth.

(g) Clean the repaired area with a clean cloth moistened with Isopropyl Alcohol, Naphtha or Toluene. Allow to air dry.

(h) Brush a minimum of two coats of Zinc Chromate primer over the repaired area, allowing each coat to dry.

(i) Refer to Section 2, and paint in accordance with applicable finish specifications.

2. Class 1 damage resulting in cracks shall be repaired as follows:

(a) Stop drill crack at both ends with 3/16 inch diameter holes.

(b) Prepare a circular external patch, which will extend one inch beyond damage area from .012 or .015 aluminum.

(c) Remove all paint and primer around damage area by sanding with 400 ro 600 grit wet sandpaper.

NOTE

Sanded area must be approximately 1/2 inch wider than aluminum patch.

(d) Lightly sand entire damage area with 400 grit sandpaper until a satin finish is obtained. Mask off around damage area.

(e) Wipe damage area with a clean cloth moistened with Isopropyl Alcohol, Naphtha or Toluene. Wipe dry with a clean cloth.

(f) Mix thoroughly 100 parts (by weight) Epon VIII Adhesive with 6 parts (by weight) of curing agent "A".

NOTE

Prepare only that quantity of material that will be used in two hours.

(g) Work some of the prepared adhesive in the crack and drilled holes. Apply a thin film over sanded surface.

(h) Prepare a one-ply No. 191 glass cloth or similar scrim cloth 1/8 inch wider than aluminum patch.

(i) Apply a thin film of the adhesive on the aluminum patch and, then place cloth between patch and damaged assembly and apply sufficient pressure to assure positive contact.

(j) Using mylar or cellophane over aluminum patch, place a clamping device on patch to insure complete contact of all bonding surfaces.

(k). Remove excess adhesive with a clean cloth dampened with Naphtha or Toluene.

(l) Cure at 150° to 200°F using heat lamps or oven.

(m) Remove clamps, pressure pads, etc., and sand away remaining excessive adhesive.

(n) Brush a minimum of two coats of Zinc Chromate primer over the repaired area, allowing each coat to dry.

(o) Refer to Section 2, and paint in accordance with applicable finish specifications.

d. Repair of Class 2 damage shall be as follows:

1. Class 2 damage to upper cowl resulting in damages that extend completely through the aluminum outer skin and into the aluminum honeycomb core but without damage to the inner skin:

(a) Carefully trim out skin to a circular or oval shape with a hole saw or fly cutter removing honeycomb core completely to the opposite skin.

CAUTION

Extreme care should be taken not to damage the inner skin.

(b) Remove completely all the damaged honeycomb core.

(c) Prepare either an aluminum honeycomb core or balsa wood replacement plug as follows:

(1) If balsa wood is used, fabricate plug so that the grain will be perpendicular to the skins.

(2) Lightly sand balsa plug with 400 grit sandpaper and wipe off dust with a clean cloth.

NOTE

Do not touch the bare surface with bare hands after sanding. If the balsa wood is cut too short, the distance between shall be shimmed up until positive contact is made with all surfaces.

(3) Wrap balsa plug in clean waxed paper until ready for use.

(4) When aluminum honeycomb plug is used, the core shall be the approximate density of the original core.

(5) Cut aluminum honeycomb plug so that the top edge will be even with the adjacent skins and completely fills the damaged area.

(d) Remove all paint and primer (approximately 1-1/2 inches larger in diameter than cutout) from around damage area with either 400 or 600 grit wet sandpaper.

(e) Mask around sanded area and cutout area.

(f) Mix thoroughly 100 parts (by weight) Epon VIII adhesive with 6 parts (by weight) of curing agent "A".

NOTE

Prepare only that quantity of material that will be used in two hours.

(g) If balsa plug is being used, spread the adhesive lightly over all surfaces. If aluminum plug is being used, brush or trowel adhesive on the internal side of the existing skin and where the plug will make contact with core.

(h) Position balsa wood or aluminum plug into place.

(i) Prepare a circular external patch, which will be one inch larger than plug hole, from .012 or .015 aluminum.

(j) Prepare a one-ply circular No. 191 glass cloth or similar scrim cloth 1/8 inch larger than plug hole.

(k) Apply a thin film of adhesive over sanded surface and place the No. 191 glass cloth or similar cloth over the plug.

(l) Clean the bond surface of aluminum patch and coat with adhesive.

(m) Assemble patch over glass cloth and plug and apply sufficient pressure to assure positive contact.

NOTE

Care should be taken to insure that plug and glass cloth remain in place.

(n) Using mylar or cellophane over aluminum patch, place a clamping device on patch to insure complete contact of all bonding surfaces.

(o) Remove excessive adhesive with a clean cloth moistened with Naphtha or Toluene.

(p) Cure at 150° to 200°F using heat lamps or oven.

(q) Remove clamps, pressure pads, etc., and sand away excessive adhesive.

(r) Brush a minimum of two coats of Zinc Chromate primer over the repaired area, allowing each coat to dry.

(s) Refer to Section 2, and paint in accordance with applicable finish specifications.

2. Class 2 damage to upper cowl resulting in damage which extends completely through the fiberglass inner skin and into aluminum honeycomb core but without damage to the outer aluminum skin.

(a) Carefully trim out skin to a circular or oval shape with a hole saw or fly cutter removing aluminum honeycomb core completely to opposite skin.

(b) Prepare a balsa wood or aluminum honeycomb plug as stated in step 1. (c) above.

(c) Sand undamaged fiberglass skin lightly approximately 2 inches out from around the hole.

CAUTION

Do not sand through fiberglass skin.

(d) Prepare two No. 181 glass fabric patches, 1/8 inch larger than hole diameter.

(e) Mix thoroughly 100 parts (by weight) Epon 828 and 10 parts (by weight) Diethlenetriamine (DTA).

(f) Coat plugs with Epon 828 and DTA mix as described in step 1. (g).

(g) Impregnate the two No. 181 glass patches with mixture to the content of approximately 50% and assemble patches over plug.

NOTE

Smooth out all wrinkles.

(h) Prepare one No. 181 glass fabric patch large enough to cover sanded area and impregnate with mixture.

(i) Assemble third patch over the two previous layers and remove all wrinkles as before.

(j) Using mylar or cellophane, cover patches and apply a clamping device.

(k) Cure assembly at 150° to 200°F for approximately 90 minutes.

(l) Remove clamps, pressure pads, etc., and sand smooth to original contour.

3. Class 2 damage to side cowl which extends completely through aluminum skin and the aluminum honeycomb core shall be repaired as follows:

(a) Repair damage as described under step 1. (d), except use Bloomingdale's HT-424 or Narmco's Metlbond 302 adhesives.

(b) The cure time for adhesives described in step (a) above, will be a minimum of 5 hours.

4. Class 2 damage to side cowl which extends completely through fiberglass skin and aluminum honeycomb core shall be repaired as follows:

(a) Repair damage as described under step 2. (d), except, use either Bloomingdale's HT-424 or Narmco's Metlbond 302 adhesives.

(b) The skins shall be fabricated from either Cordo's Pyropreg AC, U.S. Polymeric's Poly Preg 502, or Narmco's 506 (color black).

(c) The reinforcement shall be 181-150 Valan.

(d) Cure time for adhesives described in step (a) above, will be a minimum of 5 hours.

e. Repair of Class 3 damage is as follows:

1. Class 3 damage to upper and side cowls resulting in damage to both aluminum and fiberglass skins having a minimum damage size of 1.0 inch or maximum damage size of 4 inches.

(a) Prepare surfaces, plugs and patches as described in step (d).
(b) Fabricate a temporary mold or block to hold the plug in place while aluminum outer skin is being repaired.
(c) Repair outer aluminum skin in accordance with step (d).
(d) Remove temporary mold or block and repair fiberglass skin in accordance with step (d), and flare in patch with existing skin contour.

f. Repair of Class 4 damage is as follows:

1. Class 4 repairs are those repairs needed to fill voids between aluminum core and skin surfaces.

(a) Drill sufficient 1/8 inch holes in the fiberglass inner skin adjacent to voids.
(b) Remove all burrs around drilled holes.
(c) Mix thoroughly 100 parts (by weight) Epon 828 with 10 parts (by weight) curing agent "D".

NOTE

Prepare only that quantity that can be used in one hour.

(d) Using a syringe or pressure gun, inject resin mix into the aluminum honeycomb cells until they become filled.
(e) Wipe off excess resin with a cloth that has been dampened with MIBK or MEK and cover the holes with masking tape.
(f) Position the structure so that both skins will be in positive contact with resin.
(g) Cure adhesive for 2 hours at 150° to 200°F.
(h) Remove masking tape and sand away excess resin.

FIBERGLASS PARTS.

The engine cowling nose caps, tail stinger, tail surface tips, fairings, heater ducts, and other parts of the aircraft are made of fiberglass.

Repairable Damage.

Damaged fiberglass parts may be repaired by the methods shown in figure 16-7. Cut and trim the area just beyond the noticeable damage. If the parts are painted, remove paint and sand clear an area at least two inches beyond the edge of the cutout. Prepare the necessary size and number of patches of glass cloth. Mix a sufficient amount of resin in accordance with the manufacturer's instructions.

WARNING

Always follow the manufacturer's mixing instructions carefully as the mixing of peroxide and cobalt together will result in a spontaneous fire.

Be sure that your hands are free from oil, grease, and dirt. Apply an even coat of resin on the sanded area. Impregnate all the glass cloth patches by laying them on a clear paper and working the resin through the fabric with a small brush. Place the larger patch over the cutout area, working out all air bubbles and wrinkles. If the cutout is large enough to cause the patch to sag, place a suitable support behind the repair area. Coat the support with automobile wax or wax paper to prevent the resin from adhering to the support. Apply a second patch over the first patch, etc., working out all wrinkles and air bubbles. After all the patches have been applied, brush the area with an even coat of resin and allow to cure. Smooth the patch area with fine sandpaper until the desired finish is obtained. Repaint the finished area with matching paint.

Damage Necessitating Replacement of Parts.

When the fiberglass parts are torn or cracked over a large area or show signs of strain through the appearance of small cracks or show signs of loss of rigidity through the use of too much pressure on the surface, then the parts shall be removed and replaced.

POLYCARBONATE AND ACRYLIC PLASTIC BONDING.

a. When it becomes necessary to bond Polycarbonate and acrylic - such as the magnetic compass base plate to the windshield - 5% solution of methylene chloride is recommended as a bonding agent. This solution can be prepared as follows:

100 parts (by weight) Methylene Chloride
5 parts (by weight) Polycarbonate powder
(such as Lexan 105 powder).

NOTE

Due to the short pot life of this solution, no more material than that which can be used in 30 minutes should be mixed.

b. Following coating of the parts to be bonded, positive contact of the mating surfaces must be made within 10 to 15 seconds. A locally manufactured tool capable of exerting 50 to 60 psi should be used to hold the bonded parts for a minimum of 4 hours.

c. Curing the bond should be accomplished by allowing the bonded parts to remain at room temperature for at least 24 hours before any stress is applied.

REPAIR OF PLASTIC WINDOW SURFACES. (See figure 16-1.)

a. Damaged window panels and windshield of this aircraft are ordinarily removed and replaced if the damage is extensive. However, certain repairs as prescribed in the following paragraphs can be made successfully without removing the damaged part from

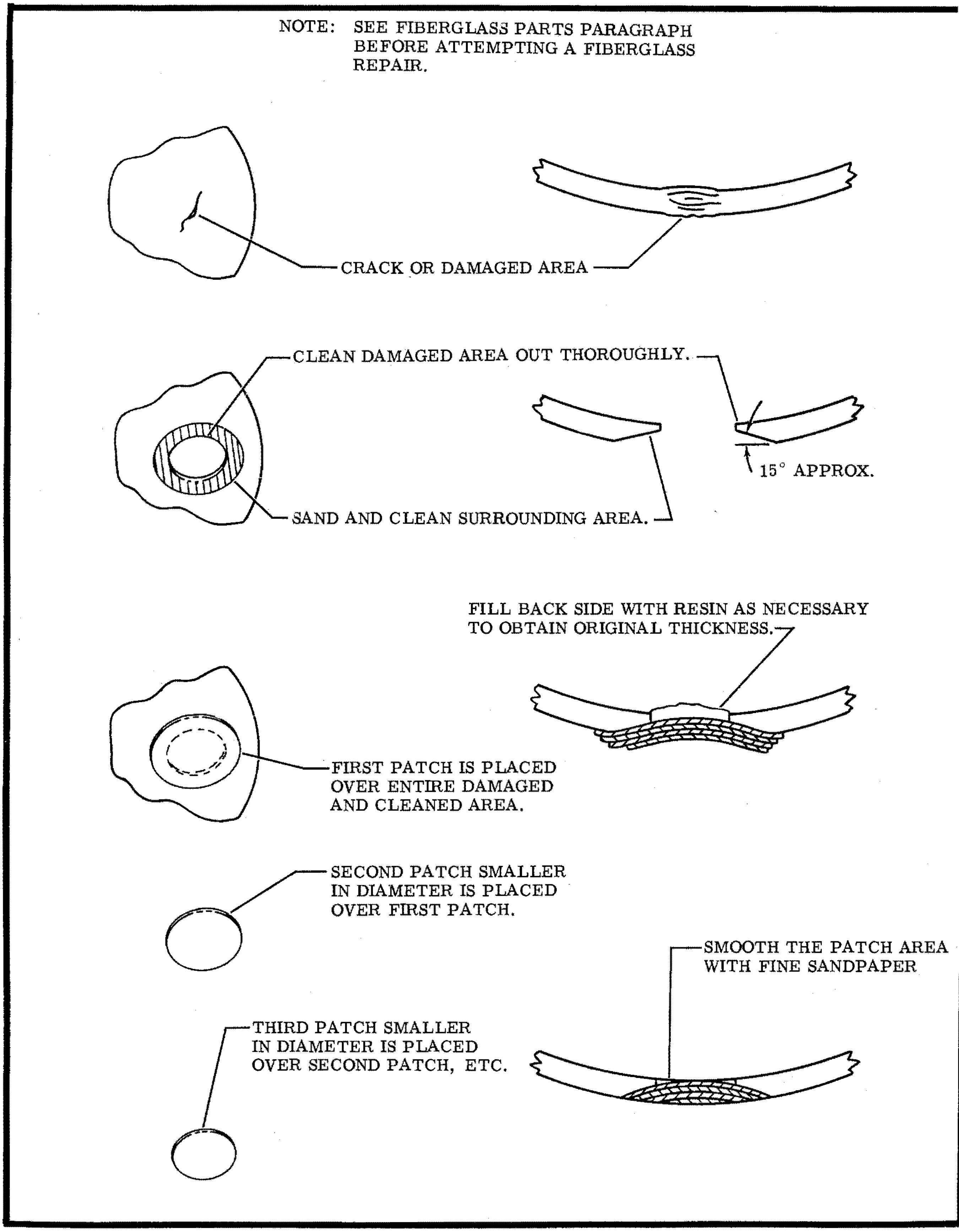



Figure 16-7. Fiberglass Repair

the aircraft. Three types of temporary repairs for cracked plastic are possible. No repairs of any kind are recommended on highly-stressed or compound curves where the repair would be likely to affect the pilot's or copilot's field of vision. Curved areas are more difficult to repair than flat areas and any repaired area is both structurally and optically inferior to the original surface.

b. When a crack appears in a panel, drill a hole at the end of the crack to prevent further spreading. The hole should be approximately 1/8 inch in diameter, depending on the length of the crack and thickness of the material.

c. Temporary repairs on flat surfaces can be effected by placing a thin strip of wood over each side of the surface and then inserting small bolts through the wood and plastic. A cushion of sheet rubber or aircraft fabric should be placed between the wood and plastic on both sides.

d. A temporary repair can be made on a curved surface by placing fabric patches over the affected areas. Secure the patches with aircraft dope, Specification No. MIL-D-5549, or lacquer, Specification No. MIL-L-7178. Lacquer thinner, Specification No. MIL-T-6094 can also be used to secure the patch temporarily.

e. A temporary repair can be made by drilling. small holes along sides of the crack 1/4 to 1/8 inch apart and lacing the edges together with a soft wire. Small stranded antenna wire makes a good temporary lacing material. This type of repair is used as a temporary measure only, and as soon as facilities are available, the panel should be replaced.

f. Scratches on transparent plastic surfaces can be removed by hand-sanding operations followed by buffing and polishing, provided subsequent instructions are followed carefully. Wrap a piece of No. 320 (or finer) sandpaper or abrasive cloth around a rubber pad or wood block. Rub the surface around the scratch with a circular motion, keeping the abrasive constantly wet with clean water to prevent scratching the surface further. Use minimum pressure and cover an area large enough to prevent the formation of the "bull's-eye" or other optical distortions.

CAUTION

Do not use a coarse grade of abrasive; No. 320 is the maximum coarseness permitted.

g. Continue the sanding operation, using progressively finer grade abrasives.

h. When the scratches have been removed, wash the area thoroughly with clean water to remove all the gritty particles. The entire sanded area will be clouded with minute scratches which must be removed to restore transparency.

i. Apply fresh tallow and buffing compound to a motor-driven buffing wheel. Hold the wheel against the plastic surface, moving it constantly over the damaged area until the cloudy appearance disappears. A 2000-foot-per-minute surface speed is recommended to prevent heating, distortion, or burns.

NOTE

Polishing can be accomplished by hand, but it will require a considerably longer period of time to attain the same result as produced by the buffing wheel.

j. When buffing is finished, wash the area thoroughly and dry it with a soft flannel cloth. Allow the surface to cool and inspect the area to determine if full transparency has been restored. Then apply a thin coat of hard wax and polish the surface lightly with a clean flannel cloth.

NOTE

Rubbing a plastic surface with a dry cloth will build up an electrostatic charge which attracts dirt particles and may eventually cause scratching of the surface. After the wax has hardened dissipate this charge by rubbing the surface with a slightly damp chamois. This will also remove the dust particles which have collected while the wax was hardening.

## RADOME REPAIR PROCEDURES.

a. Remove radome in accordance with Section 3. Repair procedures are developed with the objective of equaling as nearly as possible the electrical and strength properties of the original part with a minimum increase in weight. This can only be accomplished by repairing damaged parts with approved materials and working techniques. For convenience in presentation and for clarity in designating repair procedures to be used, damages to solid laminate radomes in this procedure shall be divided into classes according to severity, as follows:

1. Class I Repair. Surface scratches, scars or erosion not penetrating through the first ply of fabric.
2. Class II Repair. Punctures, delaminations, contaminates or fractures extending through the first ply down into the laminate but without damage to the opposite facing.
3. Class III Repair. Damage extending completely through the laminate affecting both facings.
4. Class IV Repair. Defect which does not exceed an area .5 inch square and surface has not been broken and does not occur more than twice in any 1 foot square area.
5. Class V Repair. Delamination in edge bond extending up to 1/8 inch out from drilled holes; delaminations not extending more than 1/2 inch from trimmed edges and approximately 1 inch in size.

b. Repair Techniques.

1. Class I Repair. Surface scratches, scars or erosion not penetrating through the first ply shall be repaired as follows:

(a) Clean damaged area thoroughly and carefully using a clean cloth saturated with methyl-ethyl-ketone or another approved cleaning agent.

(b) Lightly sand the damaged area, using No. 280 grit sandpaper, clean the sanded surface thoroughly, using methyl-ethyl-ketone, Specification TT-M-261. Moisture and solvents should be completely removed to prevent their inhibiting the cure of

the resin.
(c) Apply one or two coats (depending on severity of the abrasion) of the following resin mix to the abraded surface.

Composition

| | |
|---|---|
| Resin Selectron 5003 (Alternate: Hetron 92) | 100 parts by weight |
| Lupersol DDM | 0.5 to 1.5 parts by weight |
| Luperco ATC Paste | .95 to 1.05 parts by weight |
| Nuodex Cobalt | 4 to 8 drops/lb. resin |

Refer to last page of Section 16 for vendor information.

Mixing Procedure

Dissolve ATC Paste in the resin and thoroughly mix. Add Nuodex Cobalt and mix. Add DDM and thoroughly mix.

Bench Life

Approximately 30 minutes at room temperature.

(d) Over this coated surface, apply a sheet of colored cellophane or polyvinyl alcohol film extending two or three inches beyond the surface. The cellophane or PVA prevents exposure to the air and will provide a smooth surface against which the resin may cure.
(e) Tape cellophane or PVA in place and work out air bubbles and excessive resin with the hand or a rubber squeegee. Cure the resin as follows:
(1) Gel resin any surface temperature from 80° F up to a maximum of 150° F.
(2) After gelation the repaired area is cured at a surface temperature of 120 to 150° F for 30 minutes. The surface temperature is then raised to 200-230° F and maintained for 30 minutes. Heat may be obtained by heat lamps, glo rods, etc.
(f) After the resin has cured or set, remove the cellophane or PVA from the cured resin and remove any excessive resin by sanding lightly.

2. Class II Repair. Punctures, delaminations, contaminates or fractures extending through the first ply down into the laminate but without damage to the opposite facing.

Method I (Stepped Joint Method).

(a) The preferred method of removing damaged plies in accomplishing a Class II Repair is by the stepped joint method. For small damages the scarf method of repair may be used.
(b) Ascertain the extent of the damaged area by visual inspection, using a strong light source, prior to beginning repair. With aid of a straight edge or compass, outline the damaged area by scribing a rectangle with rounded corners or circle that will necessitate removal of a minimum of sound material. Extend the sides of the circle a distance in inches equal to the number of plies to be removed less one inch. (Five inches if six plies are to be removed, four inches if five plies are to be removed, etc.) Overlap should be at least one inch per ply of glass cloth.
(c) With the aid of a straight edge, use a sharp knife or other specially prepared cutter and cut along the lines scribed in the outermost ply. Use extreme care not to cut or score the underlying ply. A suggested method is to cut through the overlaying ply in a series of cuts rather than attempt to cut through the ply in one cut.
(d) Remove the cut outermost ply by inserting the knife blade under the corner and prying loose carefully. When this outermost ply is removed, scribe on the next exposed ply a similar outline except reducing the dimensions one inch in all directions. (Overlaps shall be one inch each ply.) Repeat this procedure until all the damaged plies have been removed.
(e) Lightly sand exposed plies and clean surfaces using methyl-ethyl-ketone, Specification TT-M-261, and allow to dry thoroughly preparatory to completing repair buildup.

WARNING

The sanding operation on glass cloth reinforced laminates gives off a fine dust that may cause skin irritation. Breathing of an excessive amount of this dust may be injurious, therefore, precaution as to skin and respiratory protection will be observed

(f) Cut patches from the same type of fabric as was used in the original part. Cut the fabric patches to the size of each hole. Impregnate the cloth patches with the resin mix specified under Repair Method I, paragraph (c). Sandwich each patch ply between two sheets of PVA or cellophane larger than the patch by at least two inches on all sides. (See Figures 16-3. and 16-4.) The impregnated glass cloth shall contain 45-50 percent of catalyzed resin after cellophane has been removed. (Weight of resin equal to weight of dry glass cloth comprises 50 percent resin content.) Brush a coat of resin on the surface of the scarfed plies.
(g) Fit the smallest patch in place taking care to avoid entrapping air under the patch. Smooth out all wrinkles and trim to fit. Add successive plies in a like manner the warp direction of each patch ply approximately the same as that of the original ply.
(h) Surround the patch with a bleeder and cover the entire area with polyvinyl alcohol film and secure in an air tight seal with either double-backed tape or extruded sealing tape or both. Evacuate the area under the PVA film. Sweep out entrapped air and excess resin with the motion of a rubber squeegee or a similar device. The motion of the squeegee shall be slow enough so that the air bubbles will be swept clear of the laminate by the wave, or motion of the excess resin. The air bubbles may be observed through the transparent PVA sheeting and the wiping process will continue until all air bubbles are swept past the edge of the laminate. The working or wiping of the lami-

nate will be stopped when the plies of fabric are firmly packed together. Further wiping will create air or vapor voids observable as a whitening and a loss of transparency. Should the PVA sheet be punctured during the void-free working or wiping process, the hole may be repaired with transparent tape and the air which has penetrated the bag whall be worked from the laminate. Vacuum pressure shall be maintained during the complete curing process.

(i) Cure repaired area as described under Repair Method I, paragraph (c). After the resin has cured, remove the PVA from the cured resin and remove any excessive resin by sanding lightly.

Method II (Scarf Joint Method).

(a) The scarf joint method may be used when the repair of a damaged area will require removal of an area less than three-inches in diameter.

(b) The scarf method consists of sanding out the damaged plies to a circular or oval disc shape. (See Figure 16-5.) The damaged plies will be scarfed back carefully to a distance of at least 50 times the total face ply thickness by using polisher-sander Stock No. 5130-537-3394, or by hand-sanding using No. 180 grit sandpaper. The scarfing operation will be performed very accurately to provide a uniform taper and usually requires some practice before acceptable scarfs are obtained.

(c) The glass cloth laminations for the facing repairs are prepared with the largest piece being cut to the exact shape of the outside of the scarfed area. The smallest piece is cut so that it overlaps the scarfed area by its proportionate amount, depending on the number of plies in the repair and the intermediate pieces are cut so as to have equal taper.

(d) Process the prepared area in accordance with paragraphs (f), (g), (h) and (i) under Class II Repairs, Method I.

3. Class III Repair. Damage extending completely through the laminate affecting both facings.

(a) Damages completely through the laminate shall be repaired by removing and replacing the damaged material as previously outlined under Class II Repair. Never remove inner and outer radome face ply at the same time. One facing will be completed before repair is made on the opposite facing. On solid laminate radomes, 1/2 of the damaged face plies will be removed from one side and the buildup repair completed, then repeat removal and new plu buildup procedures on opposite side.

(b) To accomplish Class III Repairs, it is necessary that the opposite side of the laminate be provided with a temporary mold or block to hold the laminate in place during the first face ply buildup.

(c) The damaged facing shall be removed and replaced as previously outlined for removing and replacing damaged plies for Class II Repair. Repeat repair procedures on the opposite facing except the cut out ply area will be larger by approximately two inches than the first ply cut out area on the opposite face repair. This will prevent the joints in the inner and outer repair area from being in the same position. (See Figure 16-6.)

4. Class IV Repair. Defects which do not exceed an area 0.5 inch square and the surface has not been broken and does not occur more than twice in any 1 foot square area.

(a) If surface is broken, sand smooth.

(b) Drill 2 to 3 No. 50 holes into the damaged skin spaced throughout the damaged area. Inject resin mix under Class I Repair, paragraph (c), with a hypodermic syringe and needle to insure contact with all surfaces and to obtain a maximum possible "wetting up" of the fractured skin.

(c) Cure as described under Class I Repair, paragraph (c).

5. Class V Repair. Delamination in edge band extending up to 1/8 inch out from drilled holes; delaminations not extending more than 1/2 inch inward from trimmed edges and approximately 1 inch in size.

(a) Work as much resin mix as possible under Class I Repair, paragraph (c), into the discrepant area. Resin may be injected into the delaminated area along the panel edges with a hypodermic syringe and needle through holes drilled with a No. 50 drill bit. Extruded Sealing Tape placed around the needle and against the panel will force the resin throughout the delaminated area.

(b) Apply pressure, if necessary, with C-clamps or vacuum blanket. Cure per paragraph (3), under Class I Repairs.

Rain Erosion Coating Application (Radome).

Rain erosion resistant coatings shall be repaired if worn away, blistered, peeled or otherwise defective. If the area is exposed to spilled or leaked oil, use only the epoxy enamel finish described later in this paragraph. Normal rain erosion coating specifications are:

MIL-C-7439, Coating Systems, Elastometeric, Rain Erosion Resistand and Rain Erosion Resistant with Anti-static Treatment for Exterior Aircraft and Missile Plastic Parts.

The repair materials are listed below, in order of their application:

Primer: Bostick No. 1007, made by B. B. Chemical Co., Cambridge, Mass. Methyl-ethyl Ketone is a suitable thinner.

Rain Erosion Resistant Coating: consists of No. 1801C Top Coat Cement and No. 983C Accelerator. Both made by Goodyear Tire and Rubber Co., Akron, Ohio.

The above items (Primer, Top Coat Cement and Accelerator) are included in Goodyear Kit No. 23-56 for brush application. For spraying, use Kit No. 23-56S which includes No. 1803C Diluting Solvent.

Goodyear Kit No. 23-57 (brushing) or No. 23-57S (spraying) Kits differ from Nos. 23-56 and 23-56S in that they include Anti-static Surfacer Cement No. 1804C and No. 983C Accelerator.

Gaco N-79 Rain Erosion Coating Kit may be used as an alternate and is manufactured by Gates Engineering Co., P. O. Box 1711, Wilmington, Delaware. This kit includes N-700-9 Top Coat Cement, N-300-9 Accelerator and N-450-9 Thinner (Methyl-ethyl Ketone or Toluene may be used as alternate).

All materials must be free of lumps, separation, gelation or dry skin on the surface of liquids. Accelerators must be stirred before mixing with other materials. Allowable age, work life, standing time at drying intervals for the materials shall all be governed by the manufacturer's instructions, where given. Damaged coatings should be removed by covering the area with a cloth pad saturated with toluene and allowing it to remain for several minutes. Remove loosened material with a stiff brush or by scraping with a dull edged metal or wooden scraper. The original surface should be exposed and remnants of the old coating removed by sanding if necessary. Rough places or pits in the fiberglass should be smoothed over by use of:

Tuf-on Putty, No. P244, manufactured by Brooklyn Varnish and Mfg. Co., Brooklyn, N. Y.

When the putty is dry, sand the surface with 180 grit or finer sandpaper, wash the fiberglass surfaces at least three times with a generous amount of xylene (TT-X-916), if not available, use toluene (TT-T-548) or Trichloroethylene Military Specification MIL-T-7003. Avoid touching the surface with bare hands before applying finish. The work must be protected from dust and other contaminants during the drying periods. A sample strip should always be coated and examined before application to the aircraft parts. Apply the primer, properly thinned, to a total dried depth of 0.001 to 0.002 inch in two or three brushed coats. Allow to dry at least five minutes between coats and 20 minutes for the last coat to dry. Mix Top Coat Cement and Accelerator in the proportions of 16 parts Cement to 1.37 parts Accelerator by volume. Apply about 4 brushed coats to a total dried thickness of 0.008 to 0.012 inch. Dry each application for a period of from ten to sixty minutes before applying the next. Air bubbles may be removed by spraying lightly with equal parts of Methyl-ethyl Ketone and toluene after each coat is applied. Protect from dust and allow the final coat to cure at room temperature until tack free. On the antenna housing only, follow with Anti-Static Surfacer Cement properly prepared by mixing with accelerator and thinning as required. Allowing a minimum of five and a maximum of 15 minutes between applications, apply three to four coats to give a total thickness of 0.001 to 0.003 inch when dry. Allow to cure to a tack free condition which requires about eight hours at room temperature or four hours if held at 65°C (149°F) +5°C. The oil resistant coating materials are:

MIL-C-8514 Wash (Pretreatment).
O-A-396 Alcohol-Ethyl, Grade III.
No. 3725 Black Enamel Epoxy, manufactured by Andrew Brown Co., Irving, Texas.
Catalyst Thinner for No. 3725 Black Enamel Epoxy, manufactured by Andrew Brown Co., Irving, Texas.

Prepare the surface as described above. Mix the Wash Primer by mixing one part of acid additive (supplied with primer) to four parts of the base primer. Thin as necessary by adding a maximum of 1 part of the Alcohol to 5 parts of well-stirred acidulated primer and again stirring. Brush or spray one coat to a cured film thickness of 0.0002 to 0.0003 inch. Allow not less than one hour nor more than four hours drying time before applying next material. The basic Epoxy Enamel and Catalyst Thinner are stirred, then mixed in equal parts and stirred well again. The mixture must then be tightly sealed and stored for one hour at 27°C (80°F). It must not be thinned or thickened but shall be discarded if not suitable for use. The mixture must be used within 24 hours or discarded. Apply in three spray coats, allowing 20 minutes between coats. For a smooth surface, sand with very fine abrasive when dry, between coats. Elevated temperature drying for 15 minutes at approximately 49°C (120°F) is permissible.

NOTE

In case of conflict between the above instructions and those supplied by the repair materials manufacturer, the latter will take precedence.

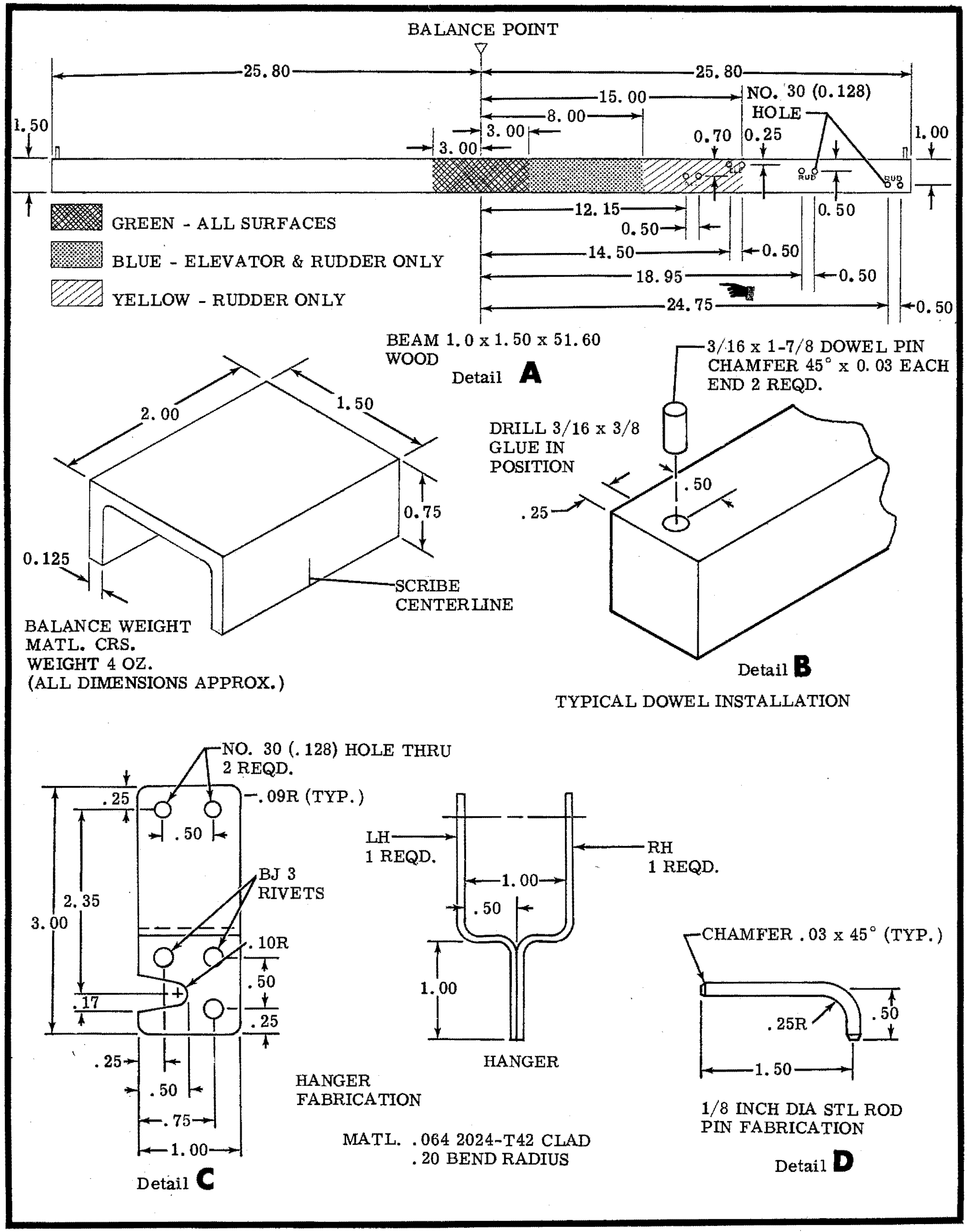
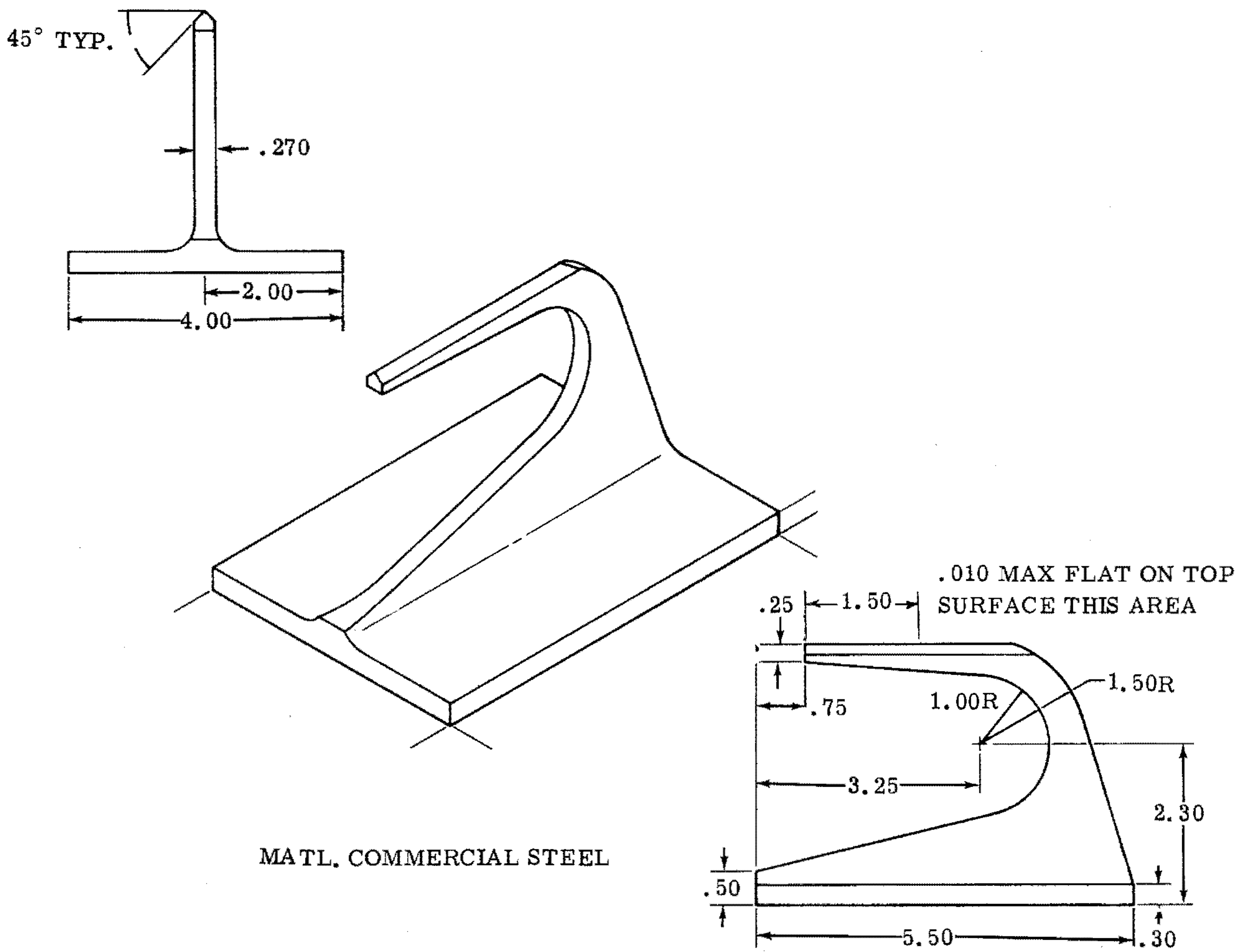



Figure 16-8. Balancing Fixture Fabrication. (Sheet 1 of 2)




FABRICATION OF BALANCE FIXTURE.

Beam.

a. Fabricate beam from wood (yellow pine) or equivalent, one by one and one-half inch stock.
b. Locate center of beam. (Should be 25.80 inches from end.)
c. Locate and drill 3/16 (.1875) x 3/8 (.375) holes .25 in from each end of beam.
d. (See Detail B.) Fabricate dowel and install in beam using glue for security.
e. Measuring from the center point of beam or 25.80 inch Station, locate and drill No. 30 (.128) diameter holes for aileron, elevator and rudder hanger locations.
f. Mark off and paint the beam as indicated for aileron, rudder and elevator positions.

Hanger.

a. Fabricate right and left hanger halves per Detail C.
b. Assemble the hanger using three MS20470AD3-5 rivets.

Pin.

a. Fabricate pin from 1/8 stl. rod per Detail D.

Balance Weight.

a. Fabricate balance weight per Detail A.

NOTE

If necessary, grind or file balance weight to obtain four (4) ounce weight. Scribe centerline after fabrication.

Mandrel.

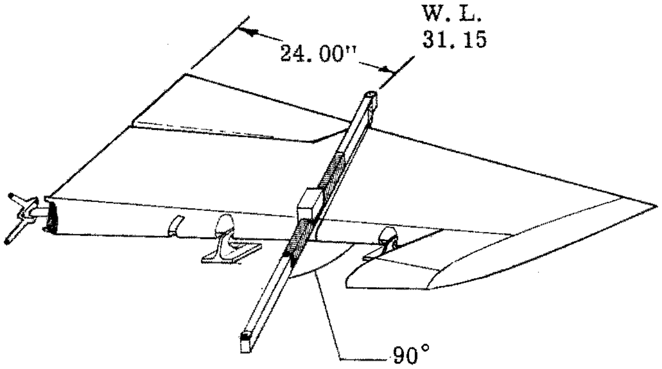
a. Fabricate mandrel in accordance with Detail E.

NOTE

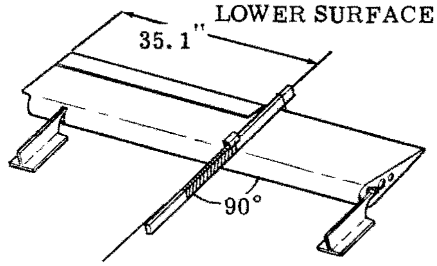
Mandrel may be fabricated from any suitable material.

Figure 16-8. Balancing Fixture Fabrication. (Sheet 2 of 2)

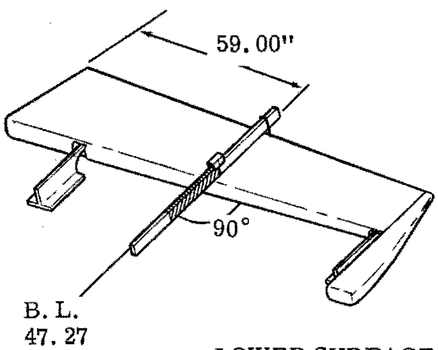
LEFT-HAND SIDE SURFACE DOWN




RUDDER:
Place balanced fixture (refer to balancing procedures) on first rib line above the trim tab on the rudder

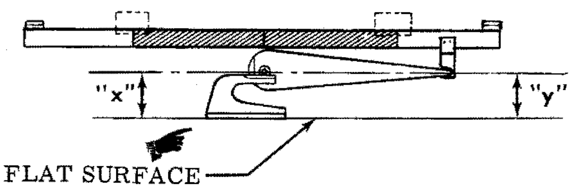



AILERON:
Place balanced fixture (refer to balancing procedures) on first rib line outboard of trim tab on the aileron




LOWER SURFACE DOWN

ELEVATOR:
Place balanced fixture (refer to balancing procedures) on first rib line outboard of trim tab on the elevator



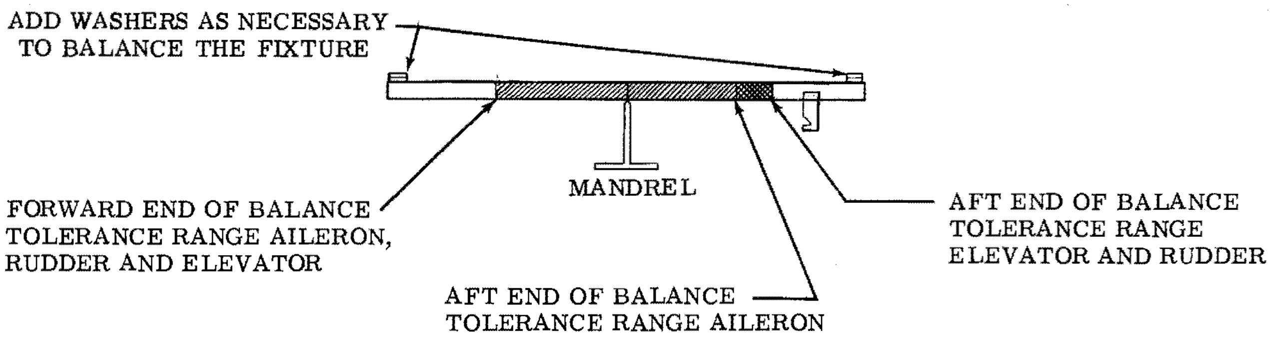

TYPICAL BALANCED CONTROL

NOTE
In a 100% static balanced condition dimensions x and y will be the same.

| SURFACE | BALANCE TOLERANCE |
|---|---|
| AILERON | +.75 TO -.75 in-lb. |
| ELEVATOR | +.75 TO -2.00 in-lb. |
| RUDDER | +.75 TO -3.75 in-lb. |

NOTE:
The above balance tolerances are provided in the event the regular balance fixture is not used. The minus tolerance is forward of hinge line and plus tolerance is aft of hinge line.

Figure 16-9. Static Balancing Aileron, Rudder and Elevator (Sheet 1 of 4)




BALANCING PROCEDURES.

It is recommended that control surface balancing be accomplished in a draft-free room/area using the balancing equipment fabricated from instructions contained in figure 16-9.

NOTE

Be certain control surface weights are installed, surfaces are clean, all foreign matter inside control surfaces removed, and control surface painted with hinge bolts installed. Surfaces which have trim tabs incorporated must have the tab secured in a streamlined position. Push-pull rods, with attaching hardware on the trim tab and rubber bellcrank, must be installed before starting balancing procedures.

BALANCING FIXTURE.

a. Install hanger assembly in the appropriate position for the surface being balanced.
b. Place mandrel on a table or other suitable flat surface and center balancing beam on mandrel at the centerline of bar.
c. Add washers to dowel pins until a balanced condition is obtained.

NOTE

When balancing fixture is in balance, it must assume a position parallel to the work surface.

RUDDER BALANCING PROCEDURE.

a. Remove rudder from aircraft in accordance with Section 7.
b. Insert hinge bolts in place and position rudder on mandrels with bolts resting on the knife edge.

NOTE

Make sure rudder control surface has freedom of movement and does not drag on mandrels and bonding wires do not interfere with movement of control.

c. Secure the rudder trim tab in a streamlined position and install associated hardware.
d. Locate the position for balancing fixture on the rudder and place balancing beam in position.

NOTE

If balancing beam position falls on a rivet line, it will be necessary to move the beam to one side to provide a flat surface for the balancing beam. The balancing beam must be perpendicular to the hinge centerline and the trailing edge of the rudder control surface must be firmly seated in the cutout of the hanger.

f. If the rudder assumes position parallel with base of mandrel and the balance weight is within the Green, Blue and Yellow range, the rudder is statically balanced.
g. If rudder assumes a position of trailing edge up, when balance weight is on aft end of range, rudder is overbalanced and needs weight removed.

NOTE

Remove overbalance by using a 1/2 inch drill to remove some of the weight.

g. If rudder control surface assumes position of trailing edge down, when balance weight is on forward end of range, rudder surface is underbalanced and needs more balance weight added.
h. The approximate amount of weight needed for underbalance may be determined by placing small amounts of loose weight in the balance weight area.
i. Additional balance weight may be added by pound-

Figure 16-9. Static Balancing Aileron, Rudder and Elevator (Sheet 2 of 4)

ing lead into existing holes of weight and staking. If holes are not available to insert added weight, it will be necessary to add 2 nutplates on each side of lightening hole and add a lead bar weight secured by two bolts.

NOTE

The rudder tip and mounting screws must be installed each time the rudder is checked for balance.

j. Install rudder in accordance with Section 7.

AILERON BALANCING PROCEDURE.

a. Remove aileron from aircraft in accordance with Section 5. (The outboard hinge is a pin.)
b. Balance the fixture for aileron control surface.
c. Insert hinge bolt in place and position aileron on the mandrels with bolt and pin resting on the knife edge.

CAUTION

Be sure bolt is inserted in correct hinge location (upper hole).

NOTE

Make sure aileron control surface has freedom of movement, does not drag on mandrels and the bonding straps are free.

d. Secure the trim tab on the left-hand aileron in a streamlined position and install associated hardware.
e. Locate the position for balancing fixture on the aileron and place balancing beam into position.

NOTE

If balancing beam position falls on a rivet line, it will be necessary to move the beam to one side to provide a flat surface for the balancing beam. The balancing beam must be perpendicular to the hinge centerline and trailing edge of the aileron control surface must be firmly seated in the cutout of the hanger.

f. If the aileron assumes a position parallel with base of mandrel and the balance weight is within Green range, the aileron is statically balanced.
g. If aileron assumes a position of trailing edge up, when balance weight is on aft end of range, the aileron is overbalanced and needs weight removed.

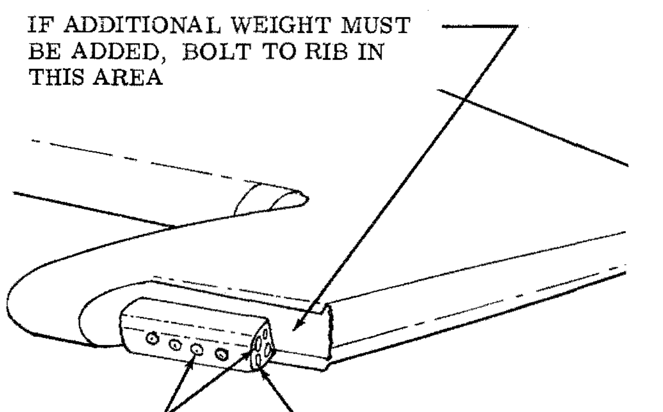
NOTE

Remove overbalance by removing weight as necessary.

h. If the aileron assumes position of trailing edge down, when balance weight is on forward end of range, aileron is underbalanced and needs more balance weight added.
i. The approximate amount of weight needed for underbalance may be determined by placing small amounts of loose weight in the balance weight area.
j. Addition of balance weight may be added by inserting lead into existing holes of the weight. If holes are not available additional weight may be added by removing existing screws and installing longer screws to attach added weight.

NOTE

The aileron balance weight must be removed from the aileron in order to remove excess weight.




RUDDER BALANCE WEIGHT

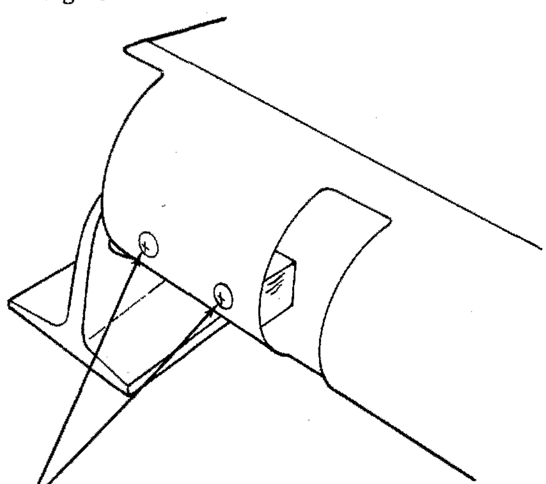



Figure 16-9. Static Balancing Aileron, Rudder and Elevator (Sheet 3 of 4)

CAUTION

Make certain bolts securing weight in place are tight before installing aileron on aircraft.

k. Install aileron in accordance with Section 5.

ELEVATOR BALANCING PROCEDURE.

a. Remove elevator in accordance with Section 6.
b. Balance the fixture for elevator control surface.
c. Insert hinge bolts in place and position elevator on the mandrels with bolts resting on the knife edge.

NOTE

Make sure elevator control surface has freedom of movement, does not drag on mandrels and bonding straps are free.

d. Secure the elevator trim tab on right-hand elevator in a streamlined position and install associated hardware.
e. Locate the position for balancing fixture on the elevator and place balancing beam into position.

NOTE

If balancing beam position falls on a rivet line it will be necessary to move the beam to one side to provide a flat surface for the balancing beam. The balancing beam must be perpendicular to the hinge centerline and trailing edge of the aileron control surface must be firmly seated in the cutout of the hanger.

f. If the elevator assumes a position parallel with base of mandrel, and the balance weight is within the Green and Blue range, the elevator is statically balanced.
g. If the elevator assumes a position of trailing edge up, when balance weight is on aft end of range, elevator is overbalanced and needs weight removed.

NOTE

Remove overbalance by using a 1/2 inch drill to remove some of the weight.

h. If the elevator assumes position of trailing edge down, when balance weight is on forward end of range, elevator is underbalanced and needs more balance weight added.
i. The approximate amount of weight needed for underbalance may be determined by placing small amounts of loose weight in the balance weight area.
j. Additional balance weight may be added by inserting lead in existing holes of the weight. If holes are not available, additional weight may be added by bolting a lead bar in the center of existing weight.

NOTE

Each time the elevator is checked for weight and balance, the elevator tip must be installed and elevator tip screws must be in the proper location (short screws toward aft end of elevator).

k. Install elevator in accordance with Section 6.

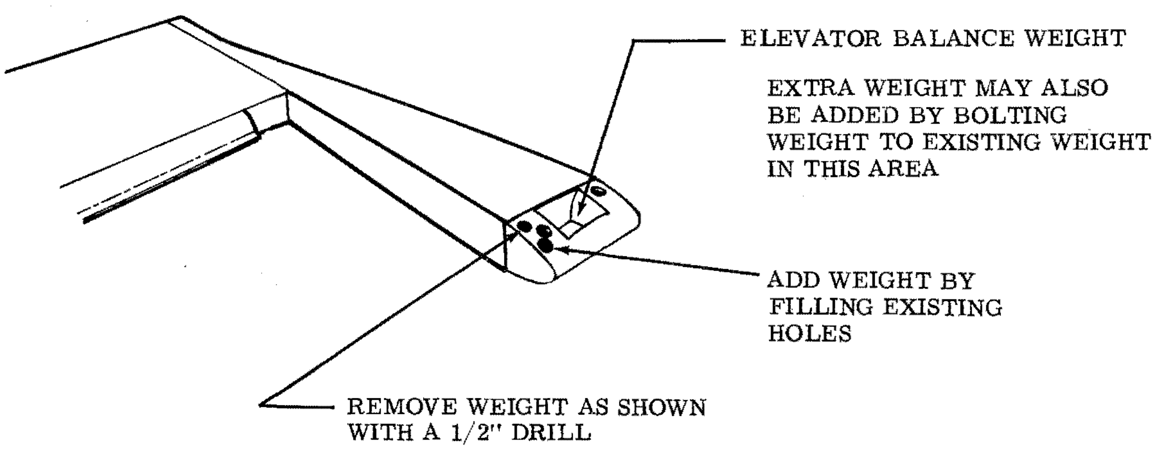



Figure 16-9. Static Balancing Aileron, Rudder and Elevator (Sheet 4 of 4)

1. FIBERGLASS
2. .016 INCH 2024 T3 ALCLAD
3. .025 INCH 2024 T3 ALCLAD
4. .025 INCH 2024 T42 ALCLAD
5. .032 INCH 2024 T3 ALCLAD
6. .032 INCH 2024 T42 ALCLAD
7. .040 INCH 2024 T42 ALCLAD
8. .020 INCH 2024 T3 ALCLAD
9. .040 INCH 2024 T3 ALCLAD
10. HONEYCOMB
11. .020 INCH 2024 T42 ALCLAD

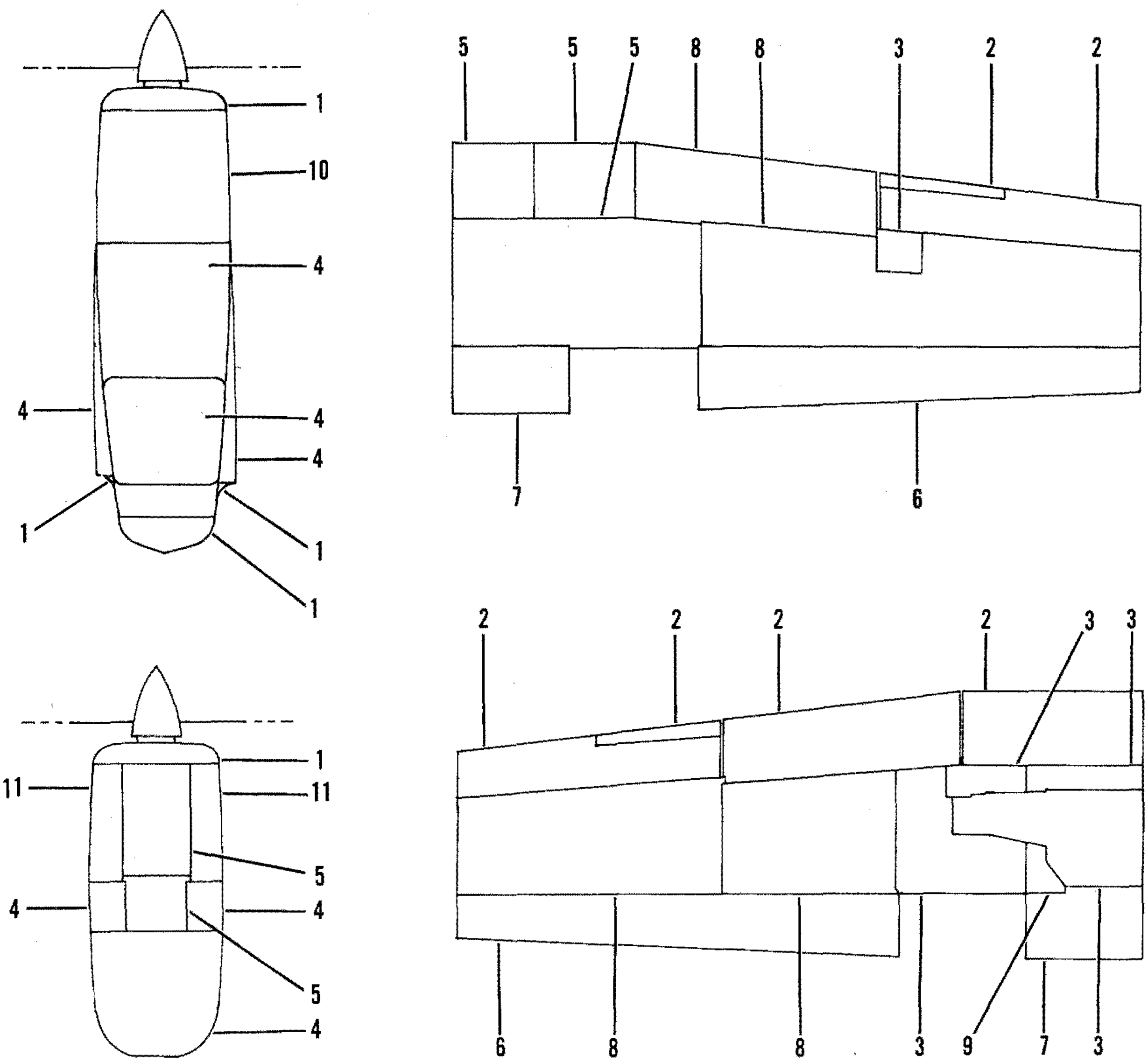


Figure 16-10. Wing Skin

1. .016 INCH 2024 T3 ALCLAD
2. .025 INCH 2024 T42 ALCLAD
3. .032 INCH 2024 T3 ALCLAD
4. .032 INCH 2024 T4 ALCLAD
5. .032 INCH 2024 T42 ALCLAD
6. .040 INCH 2024 T3 ALCLAD
7. FIBERGLASS
8. .050 INCH 2024 T42 ALCLAD
9. .063 INCH 2024 T42 ALCLAD
10. .080 INCH 2024 T42 ALCLAD
11. POLY-CARB
12. ROYALITE

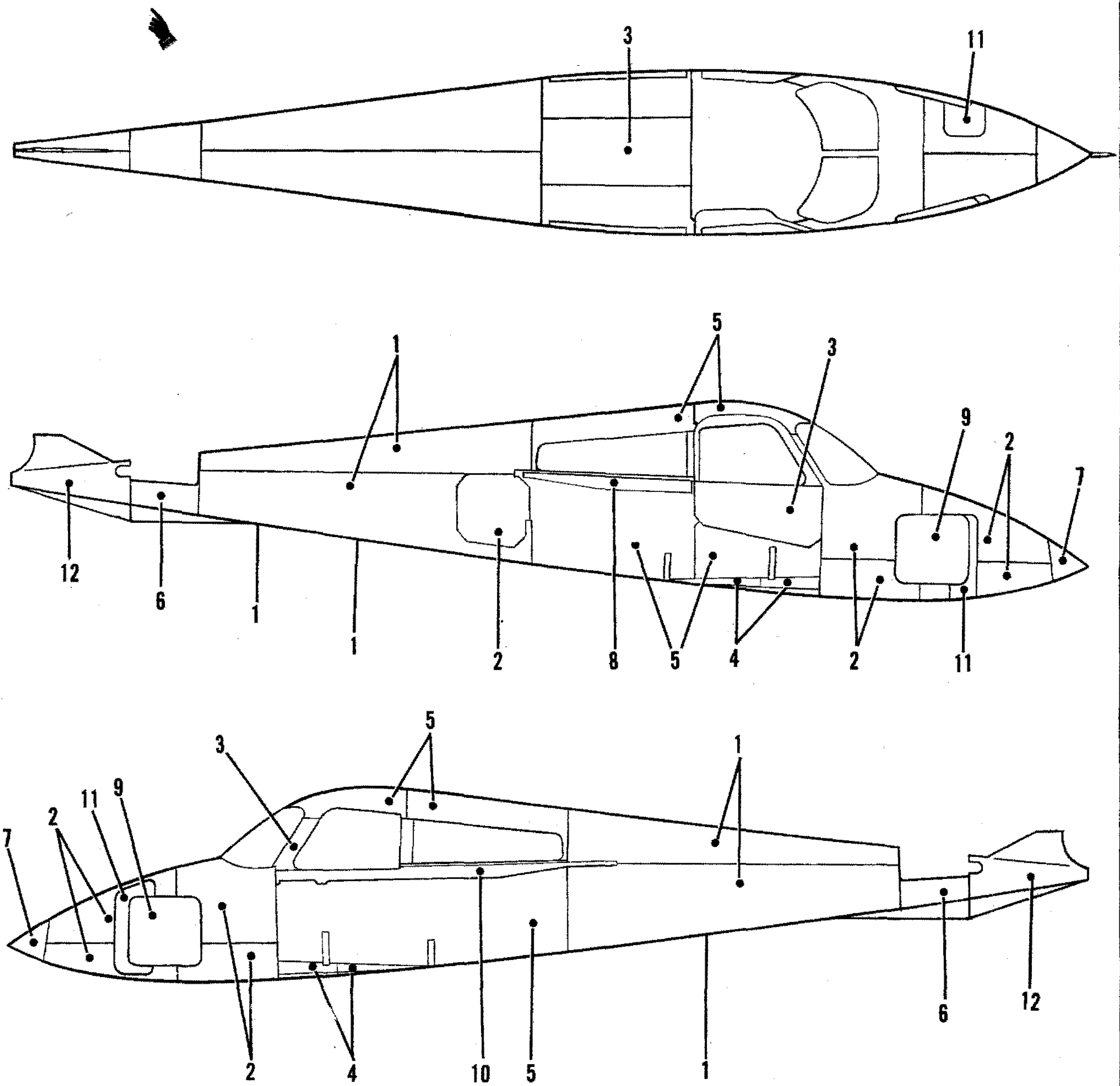



Figure 16-11. Fuselage Skin

1. FIBERGLASS
2. .016 INCH 2024 T3 ALCLAD
3. .025 INCH 2024 T3 ALCLAD
4. .025 INCH 2024 T42 ALCLAD
5. .032 INCH 2024 T3 ALCLAD
6. .020 INCH 2024 T3 ALCLAD

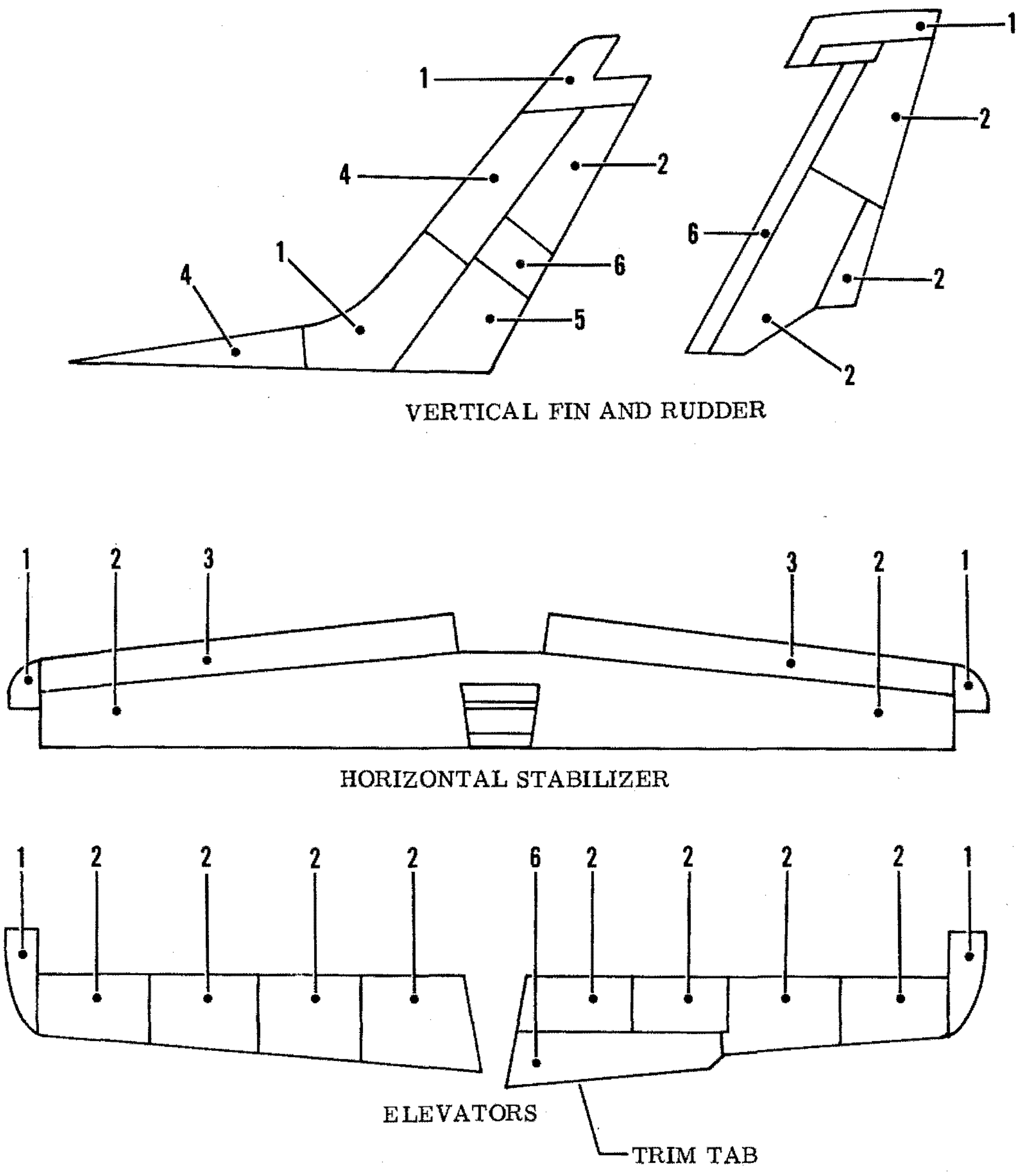
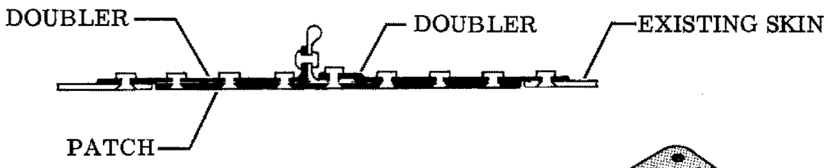
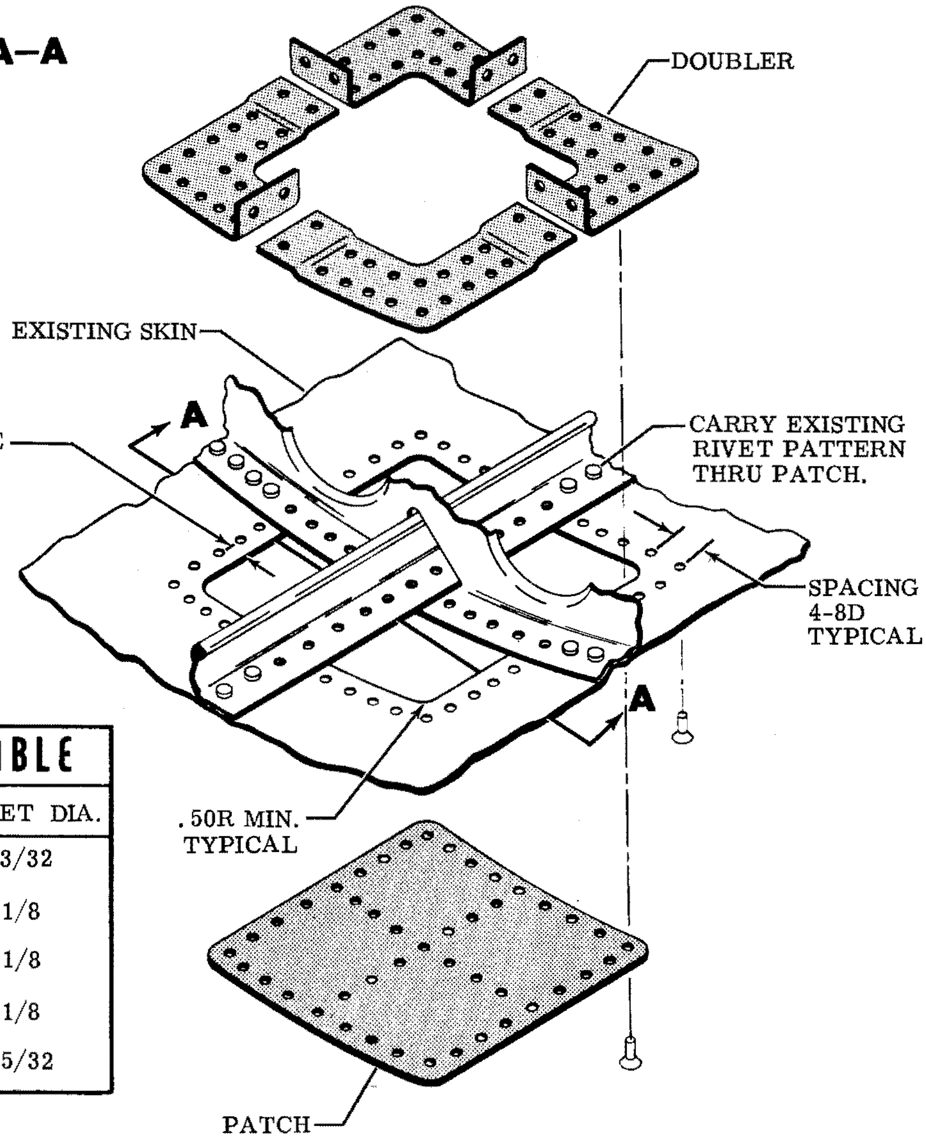



Figure 16-12. Empennage Skin




**RIVET TABLE**

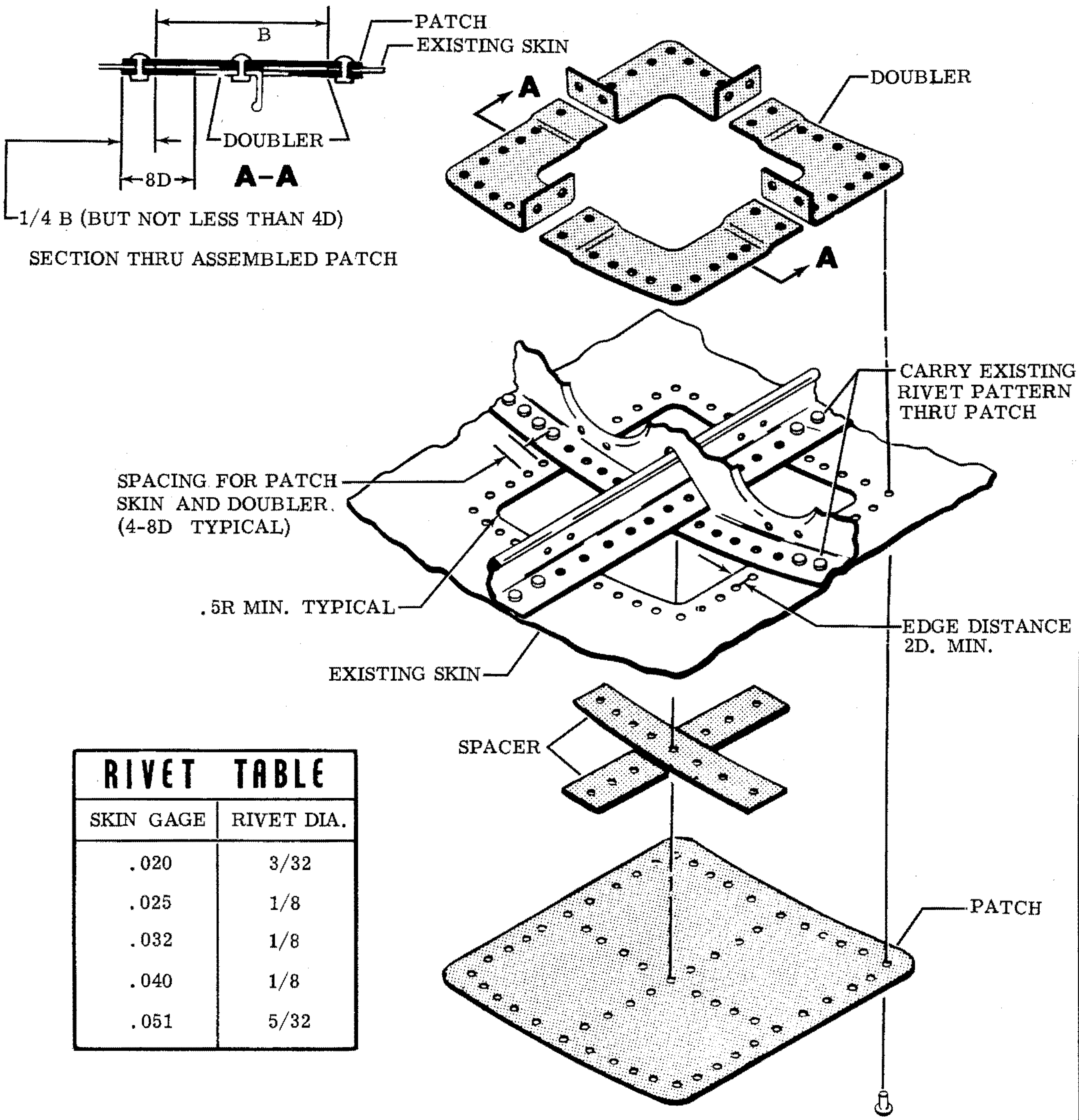
| SKIN GAGE | RIVET DIA. |
|---|---|
| .020 | 3/32 |
| .025 | 1/8 |
| .032 | 1/8 |
| .040 | 1/8 |
| .051 | 5/32 |

ORIGINAL PARTS

REPAIR PARTS

REPAIR PARTS IN CROSS SECTION

Figure 16-13. Typical Over-Structure Skin Repair (Sheet 1 of 2)




**RIVET TABLE**

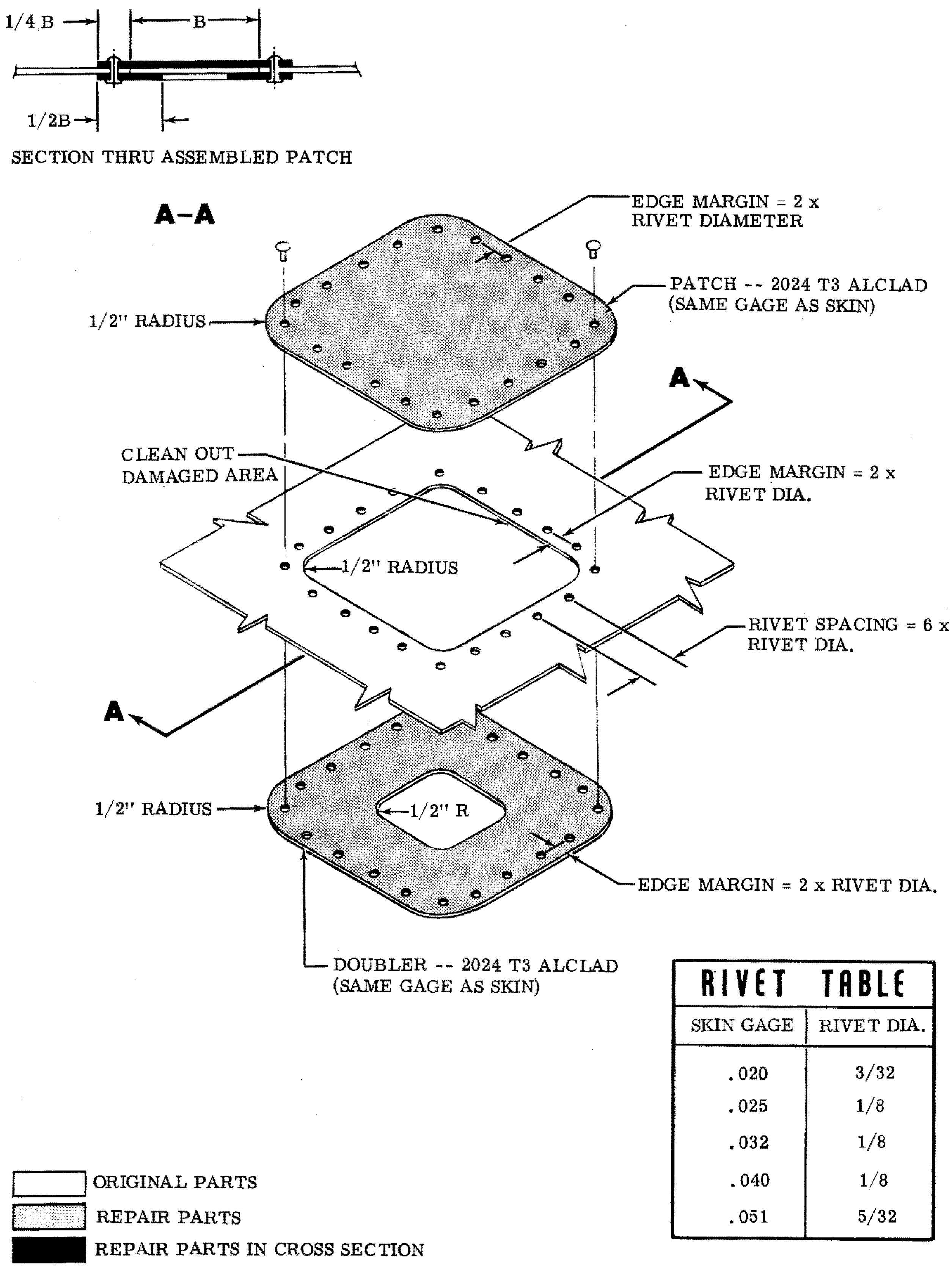
| SKIN GAGE | RIVET DIA. |
|---|---|
| .020 | 3/32 |
| .025 | 1/8 |
| .032 | 1/8 |
| .040 | 1/8 |
| .051 | 5/32 |

ORIGINAL PARTS

REPAIR PARTS

REPAIR PARTS IN CROSS SECTION

Figure 16-13. Typical Over-Structure Skin Repair (Sheet 2 of 2)




**RIVET TABLE**

| SKIN GAGE | RIVET DIA. |
|---|---|
| .020 | 3/32 |
| .025 | 1/8 |
| .032 | 1/8 |
| .040 | 1/8 |
| .051 | 5/32 |

Figure 16-14. Typical Clear-of-Structure Skin Repair

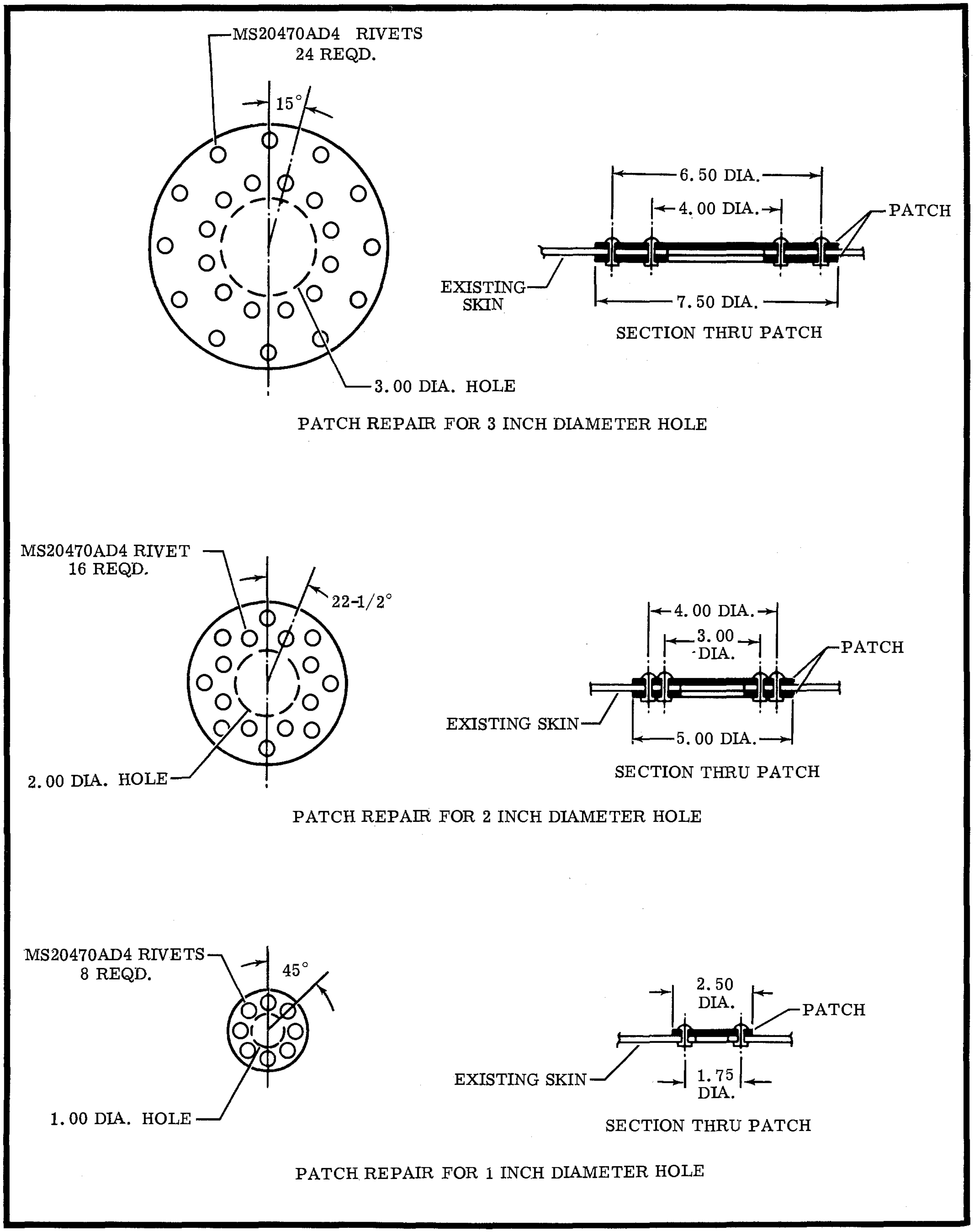
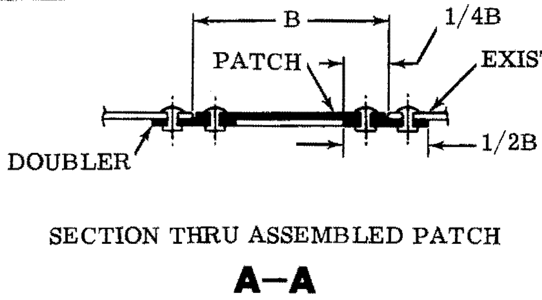



Figure 16-15. Typical Patch Repair of Circular Holes




**RIVET TABLE**

| SKIN GAGE | RIVET DIA. |
|---|---|
| .020 | 3/32 |
| .025 | 1/8 |
| .032 | 1/8 |
| .040 | 1/8 |
| .051 | 5/32 |

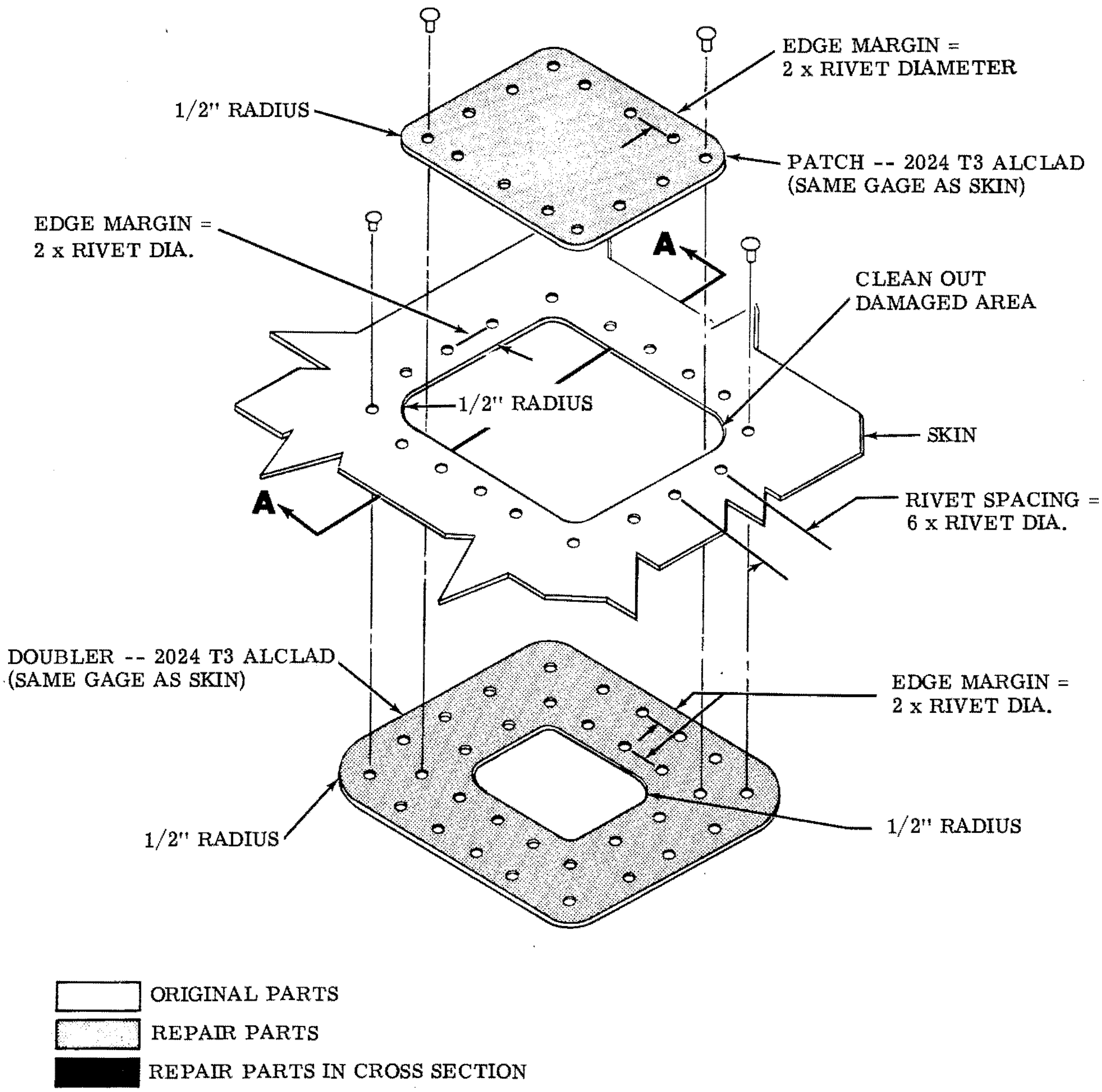



Figure 16-16. Typical Insert Patch

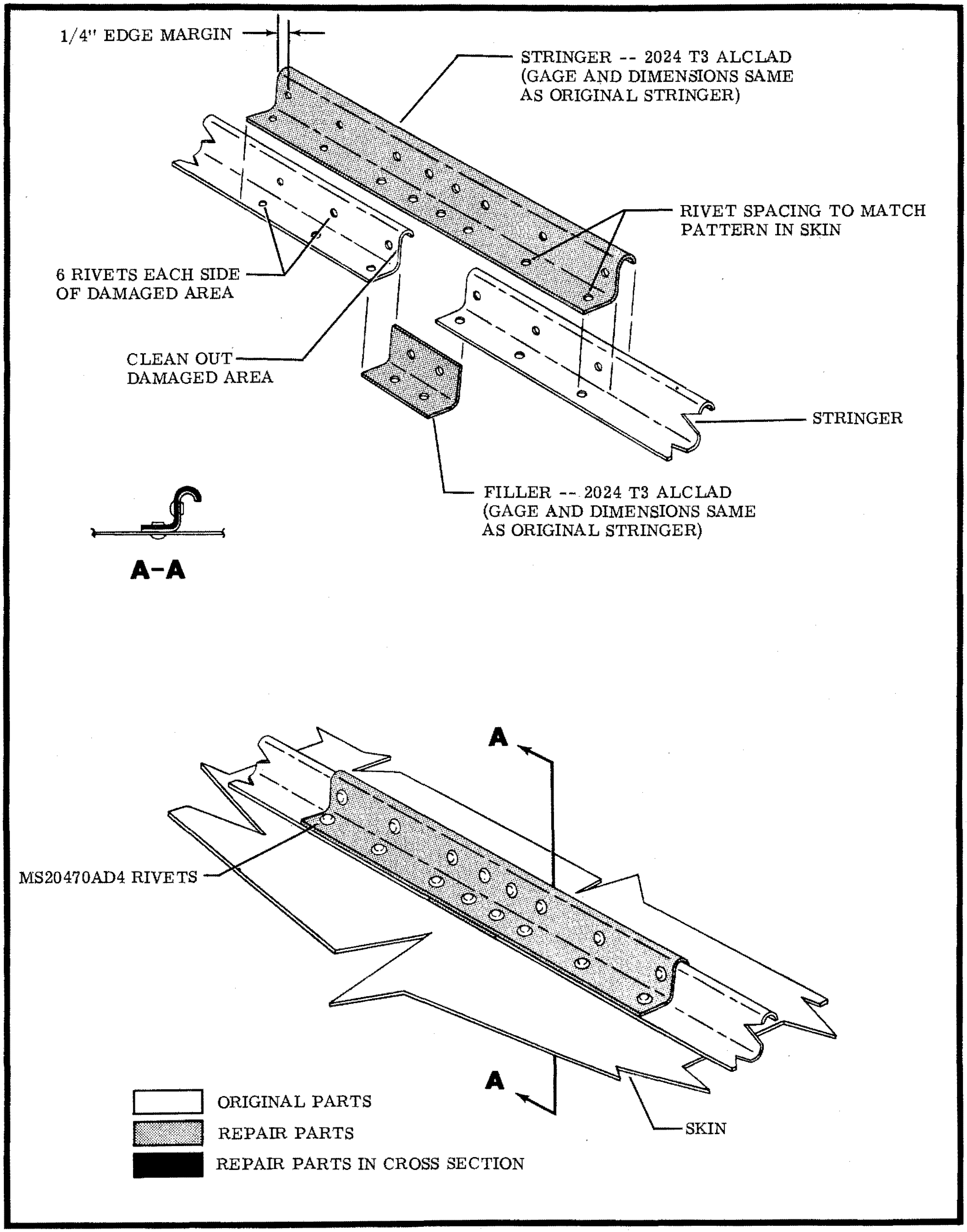



Figure 16-17. Typical Fuselage Stringer Repair (Sheet 1 of 2)

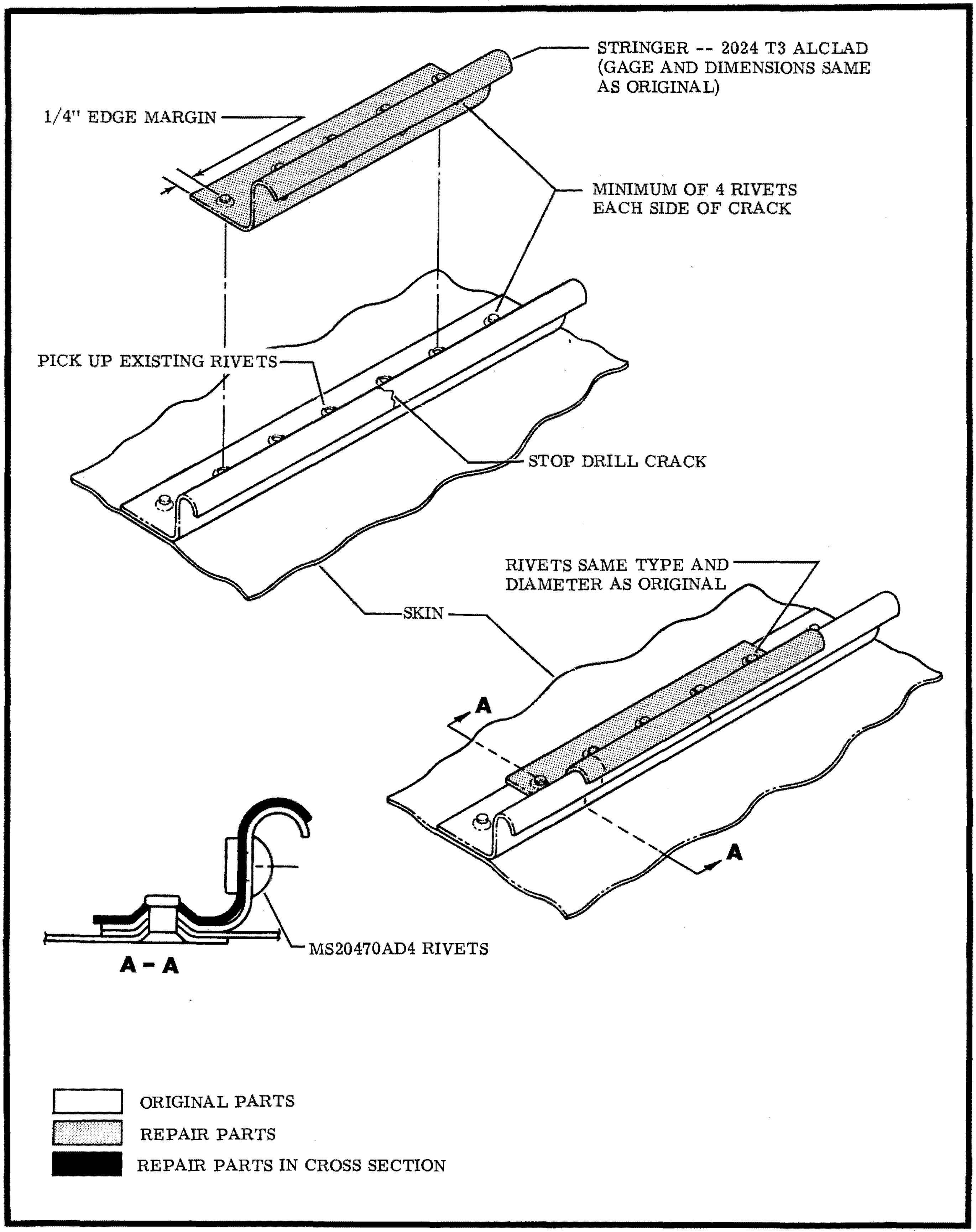



Figure 16-17. Typical Fuselage Stringer Repair (Sheet 2 of 2)

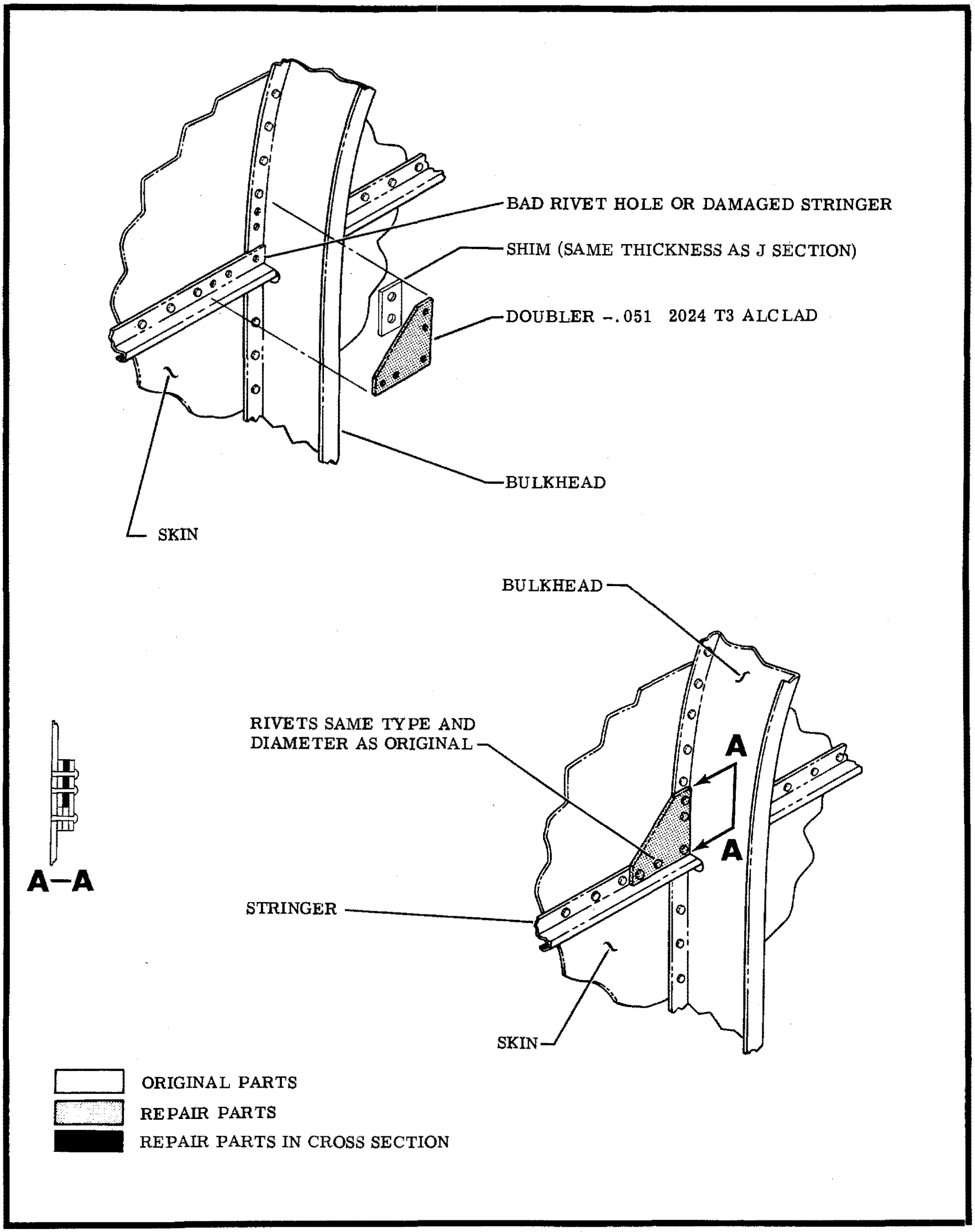



Figure 16-18. Typical Stringer to Bulkhead Repair

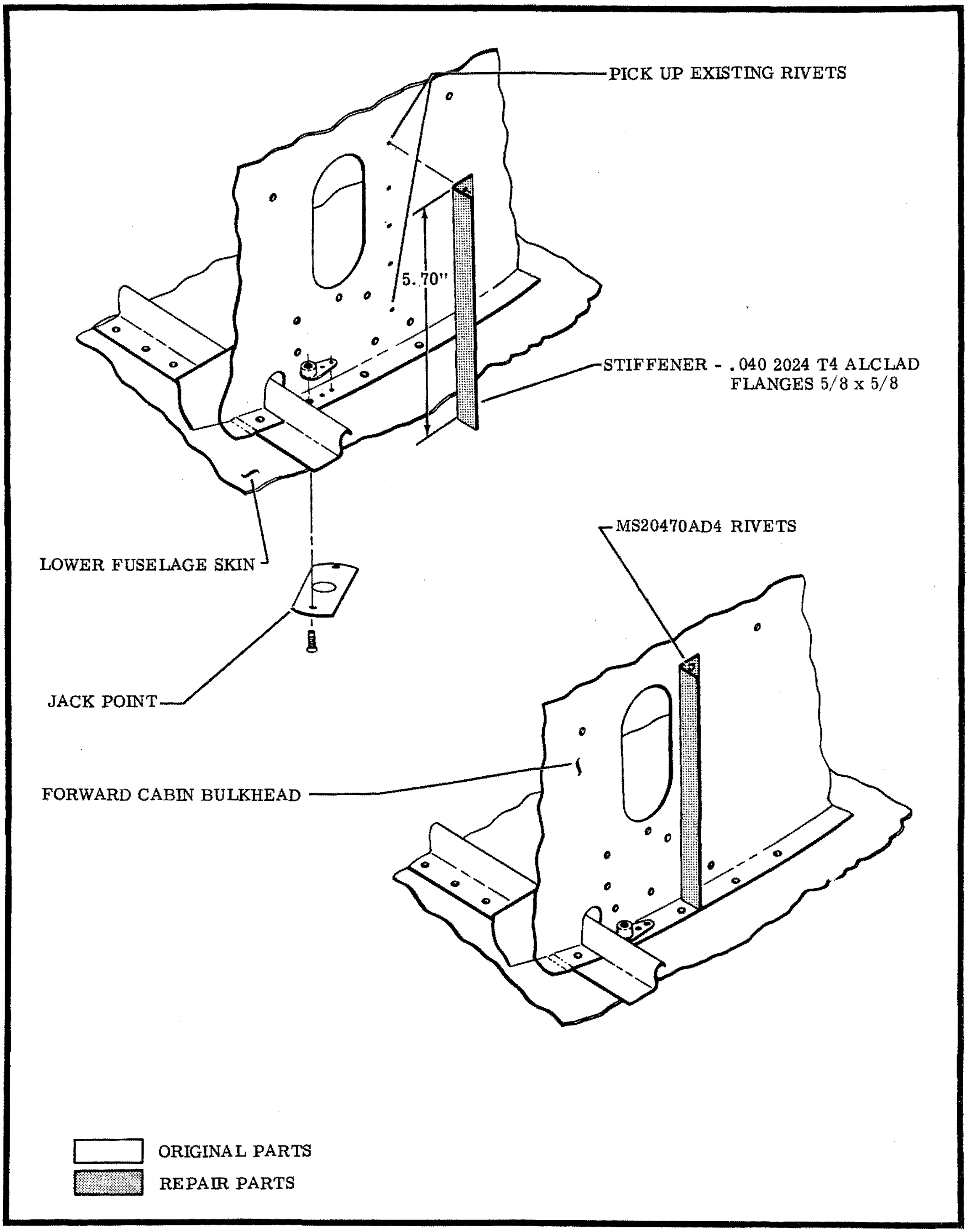



Figure 16-19. Typical Jacking Point Reinforcement (Sheet 1 of 2)

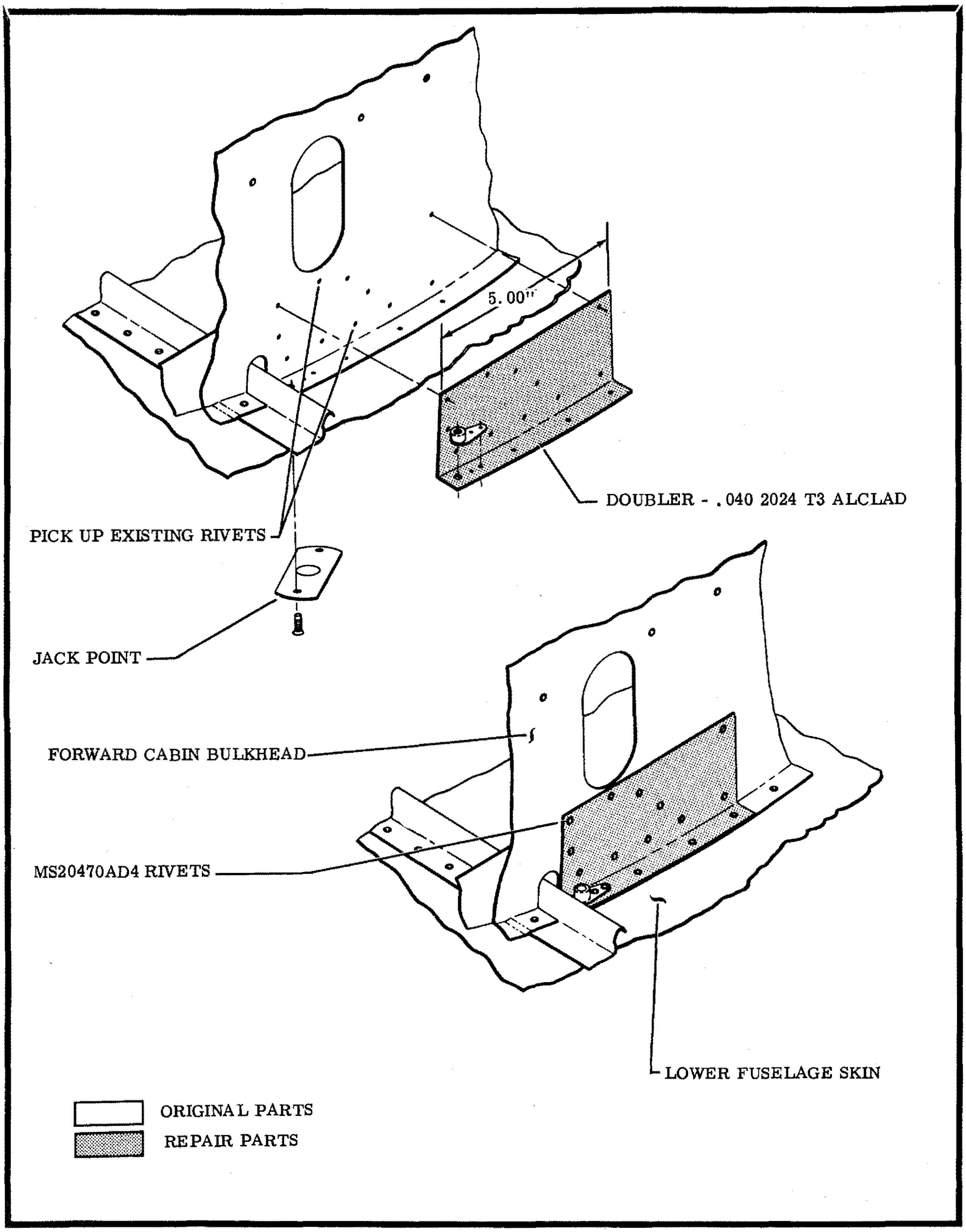



Figure 16-19. Typical Jacking Point Reinforcement (Sheet 2 of 2)

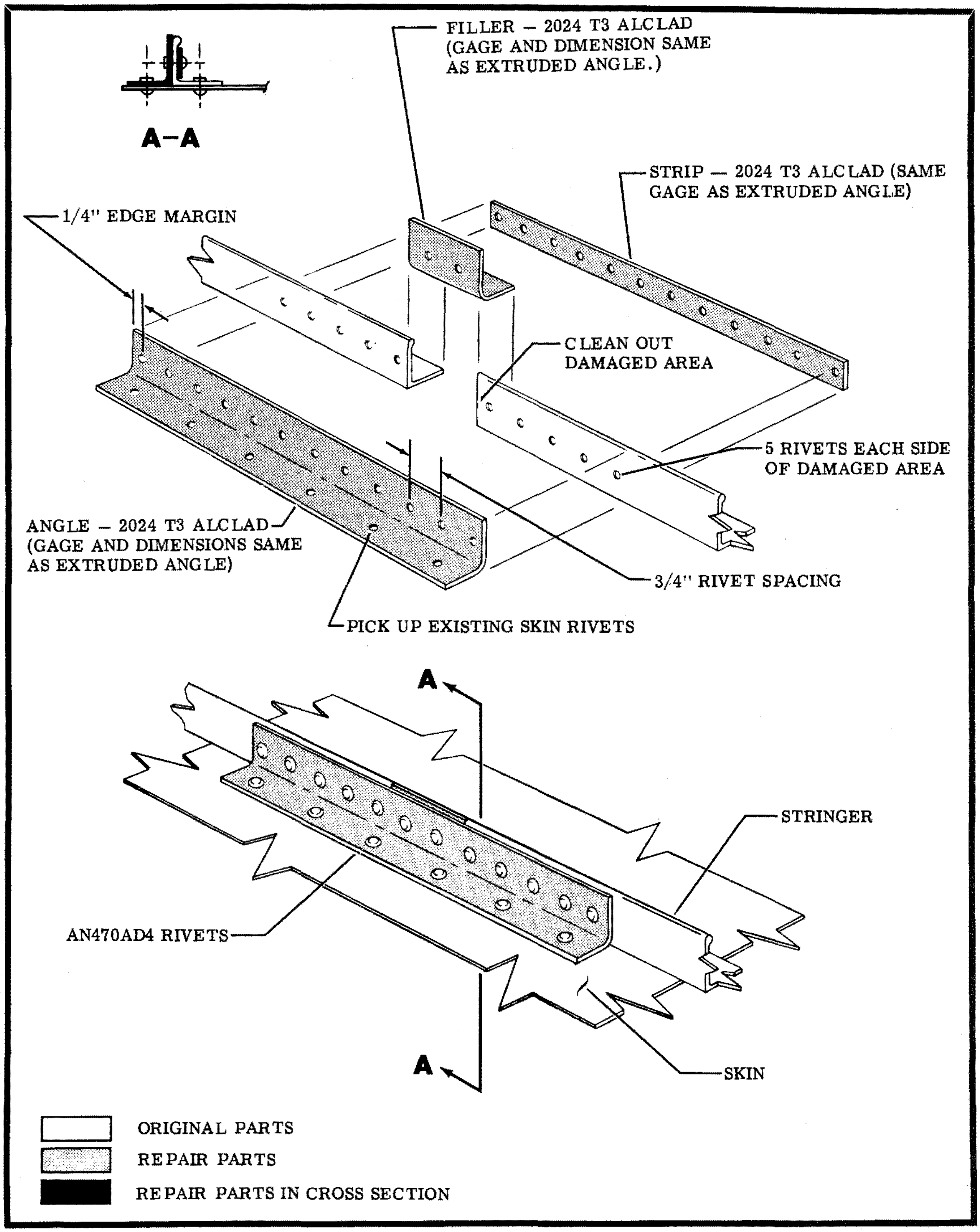



Figure 16-20. Typical Stringer Repair

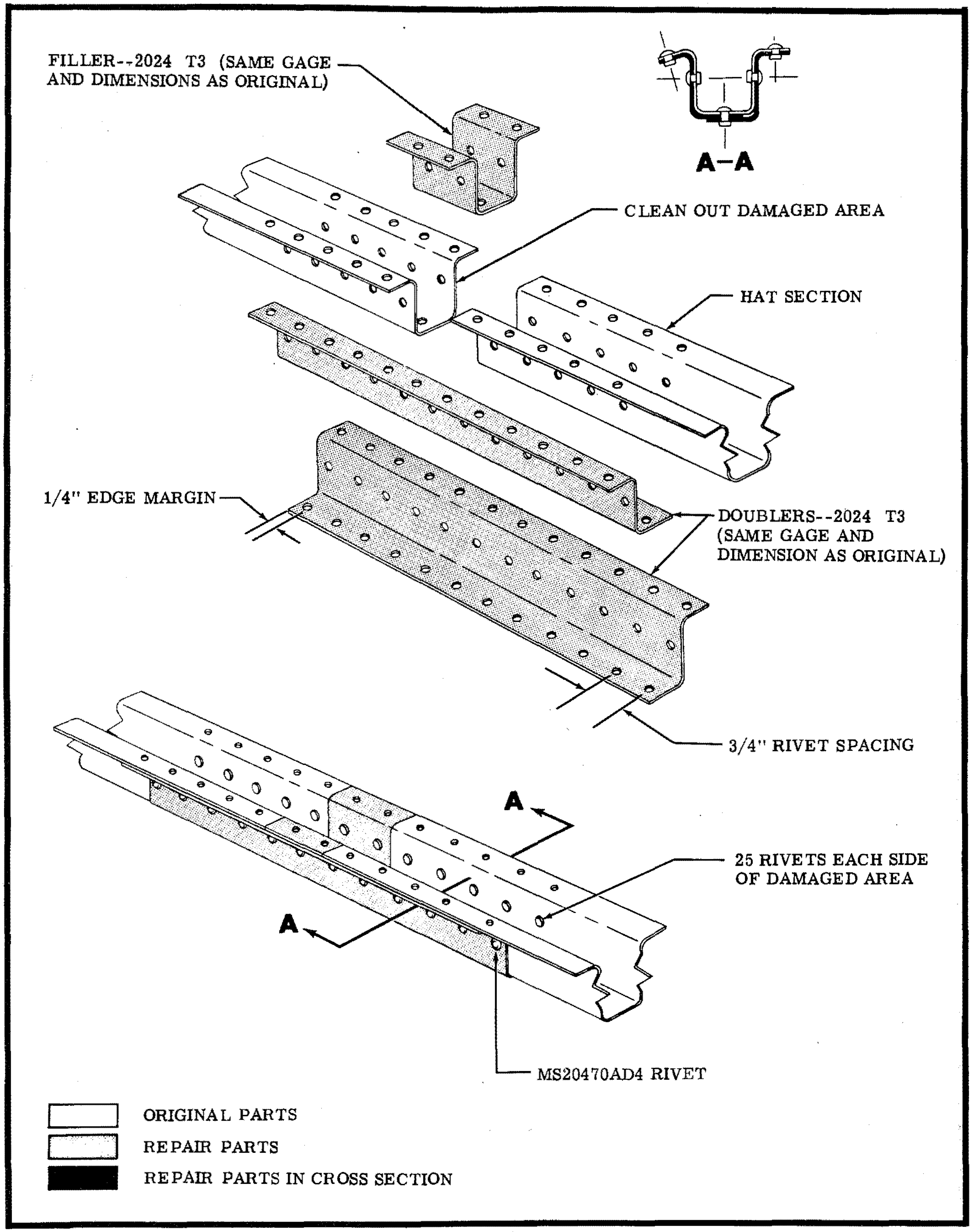



Figure 16-21. Typical Hat Section Repair

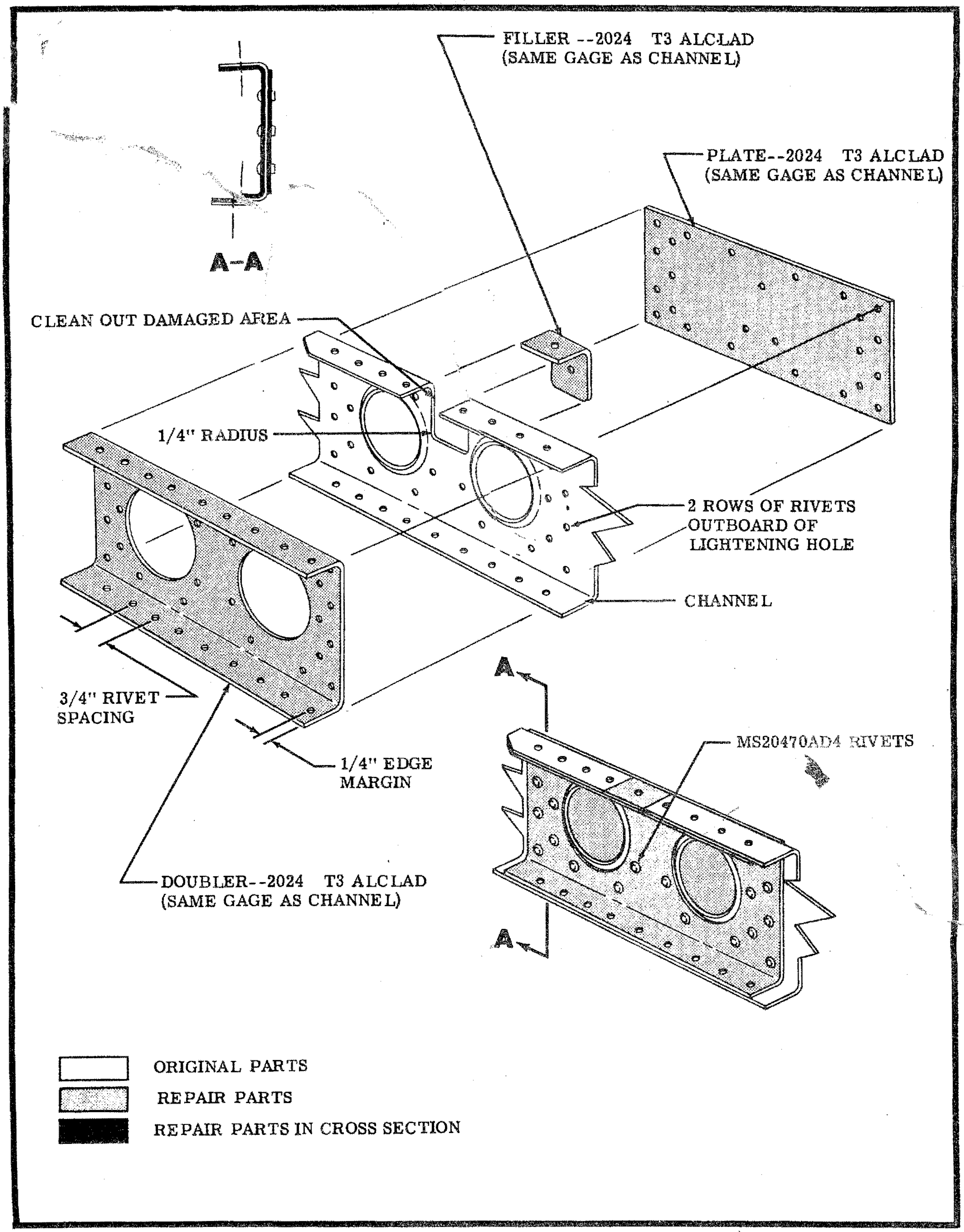



Figure 16-22. Typical Channel Flange Repair

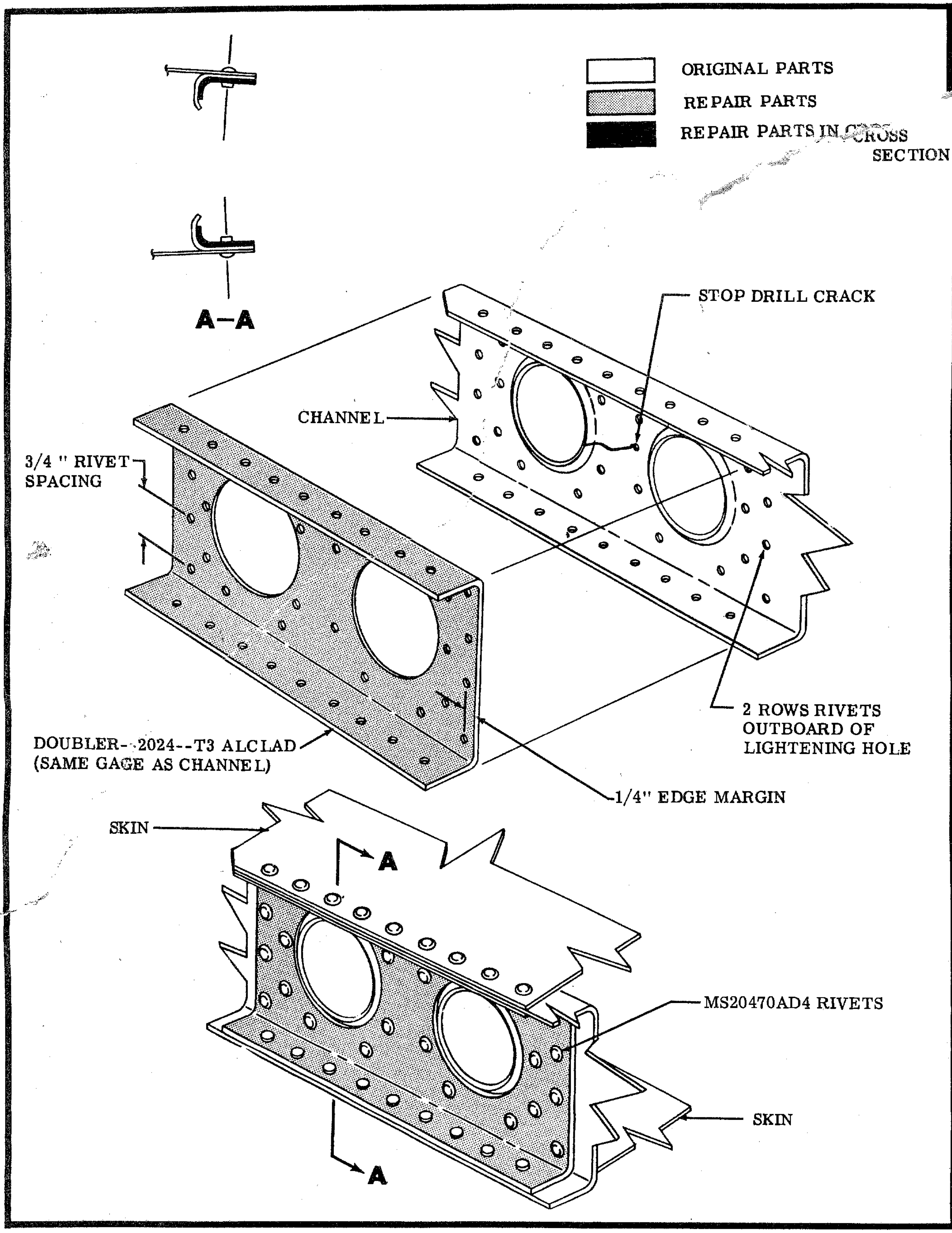



Figure 16-23. Typical Channel Repair

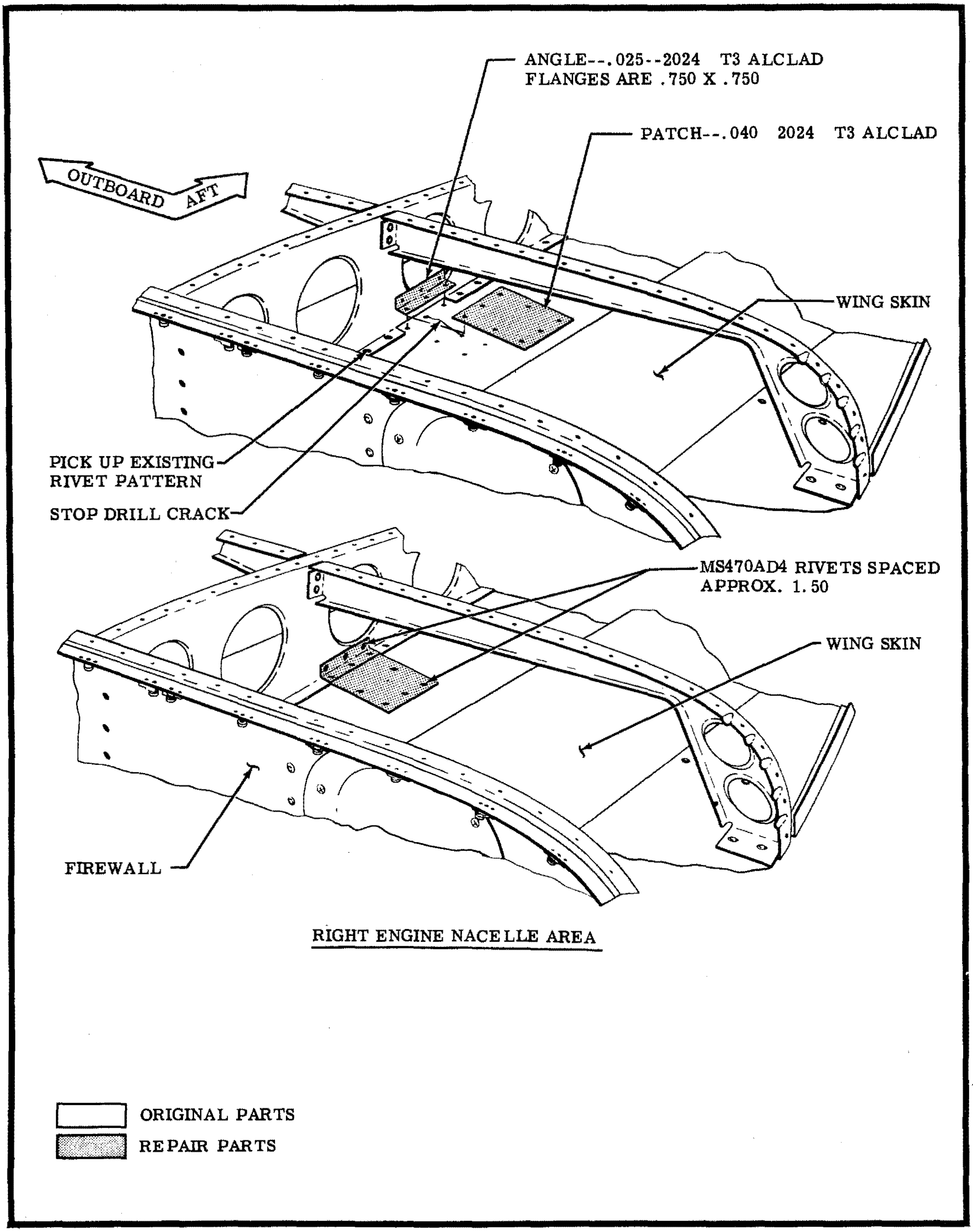
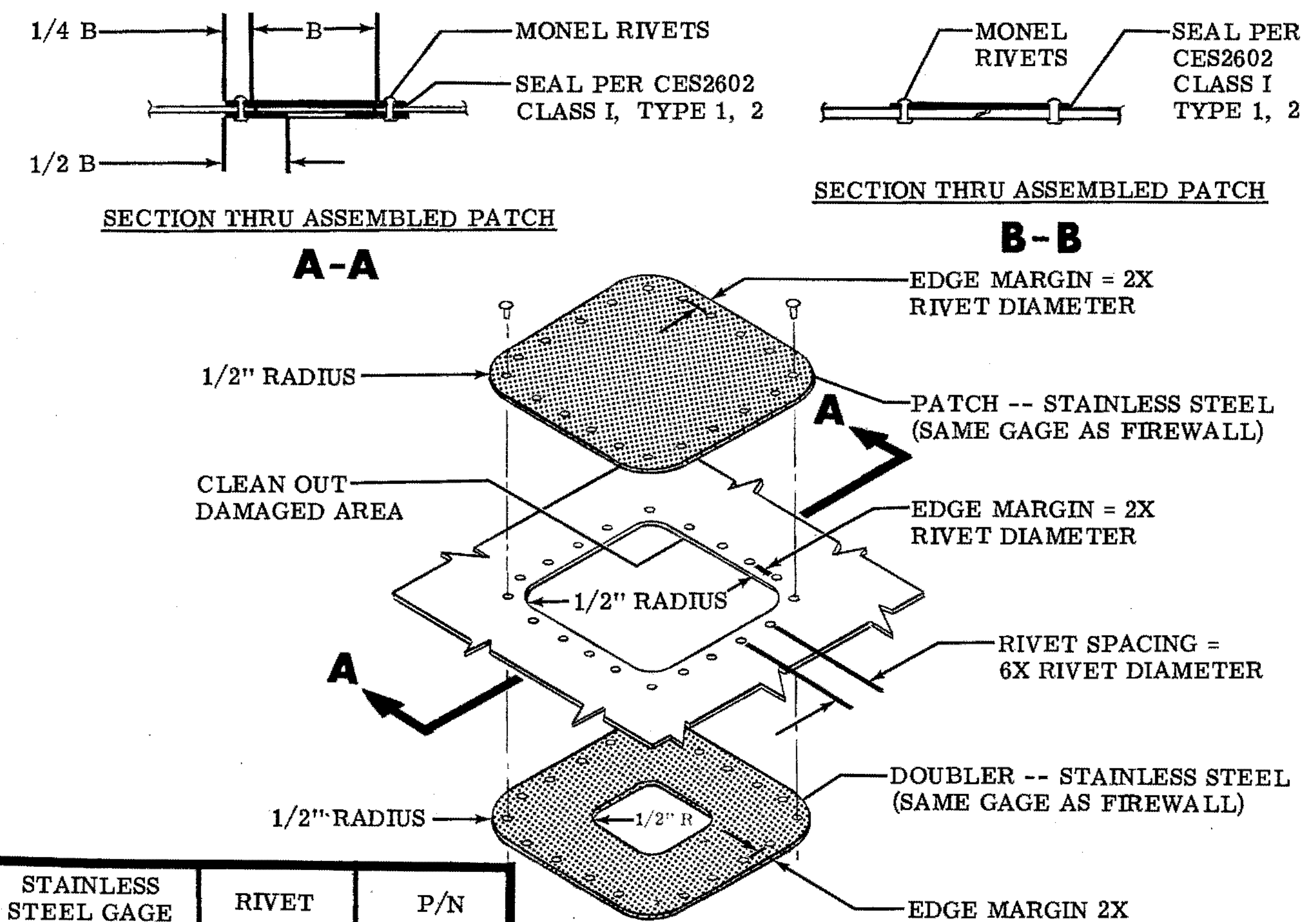



Figure 16-24. Typical Wing Skin Repair - Nacelle Area




| STAINLESS STEEL GAGE | RIVET | P/N |
|---|---|---|
| .016 | 3/32 (BLIND) | MS20435M<br>MS20427M<br>NAS1398M |

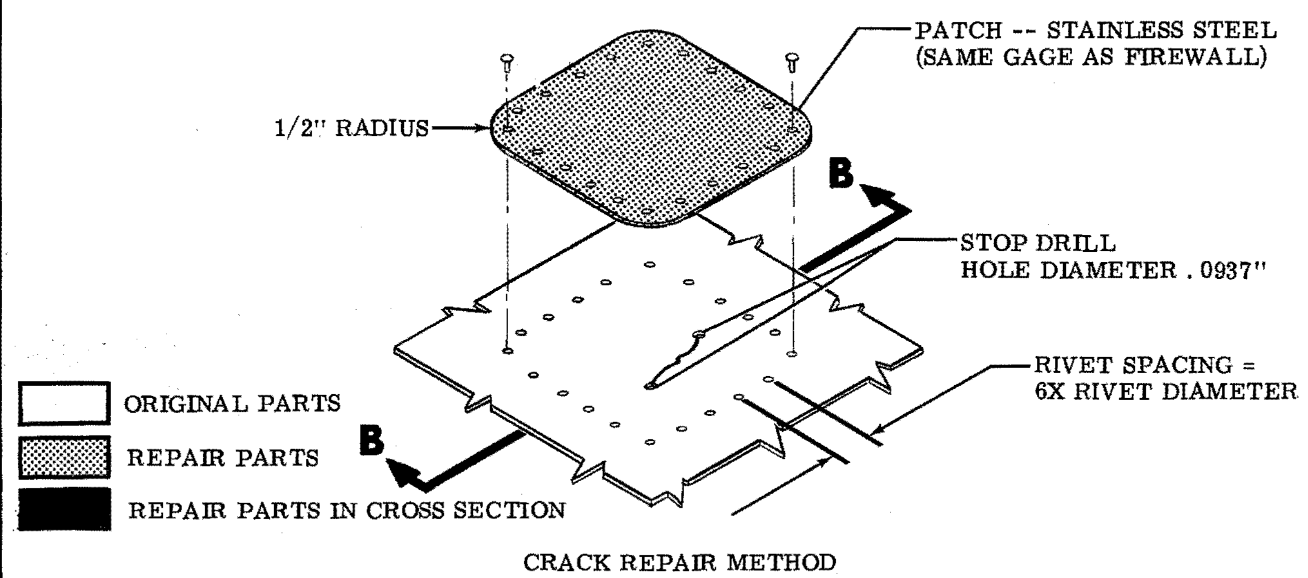
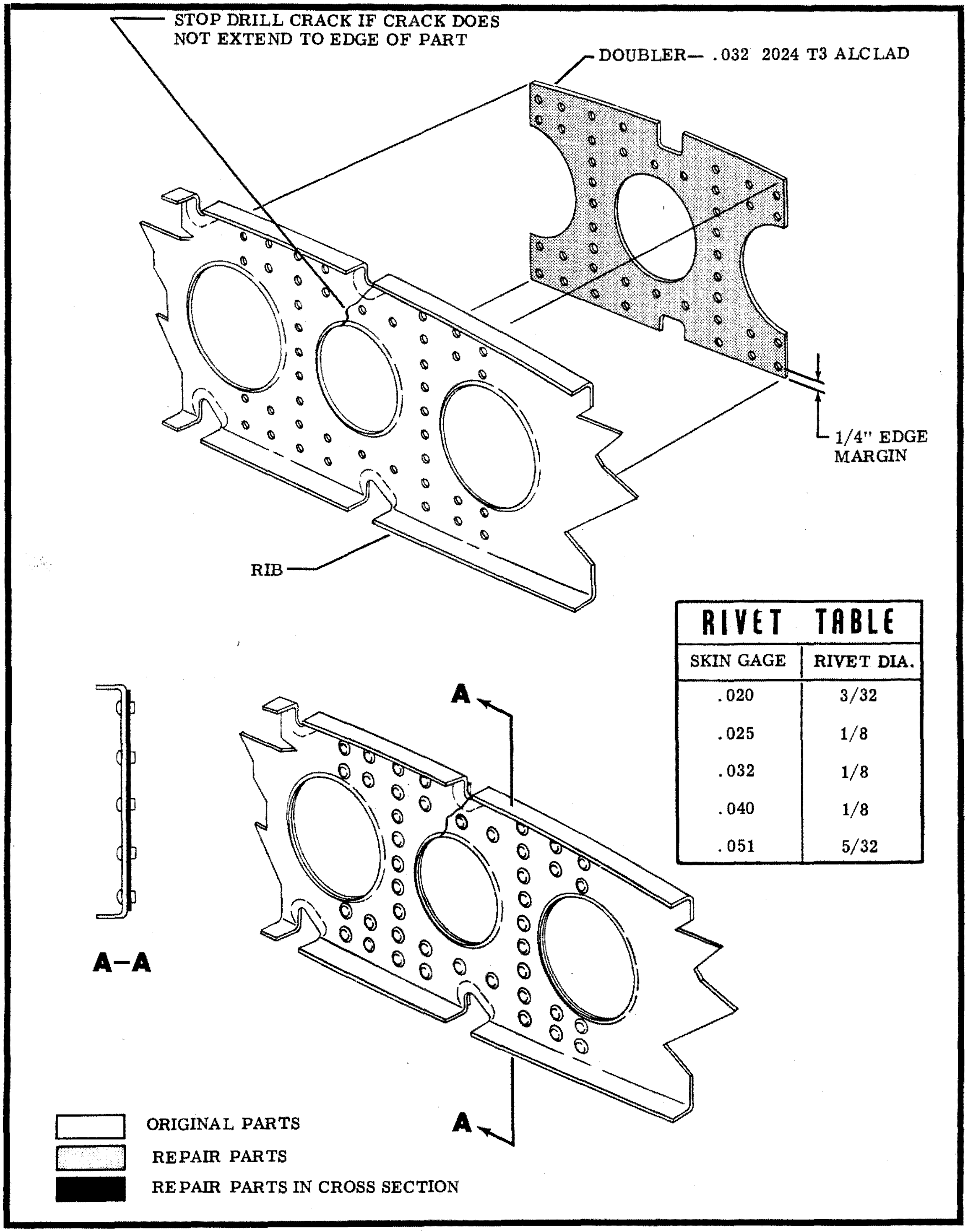



Figure 16-24A. Repair of Firewall




**RIVET TABLE**

| SKIN GAGE | RIVET DIA. |
|---|---|
| .020 | 3/32 |
| .025 | 1/8 |
| .032 | 1/8 |
| .040 | 1/8 |
| .051 | 5/32 |

Figure 16-25. Typical Rib Web Repair

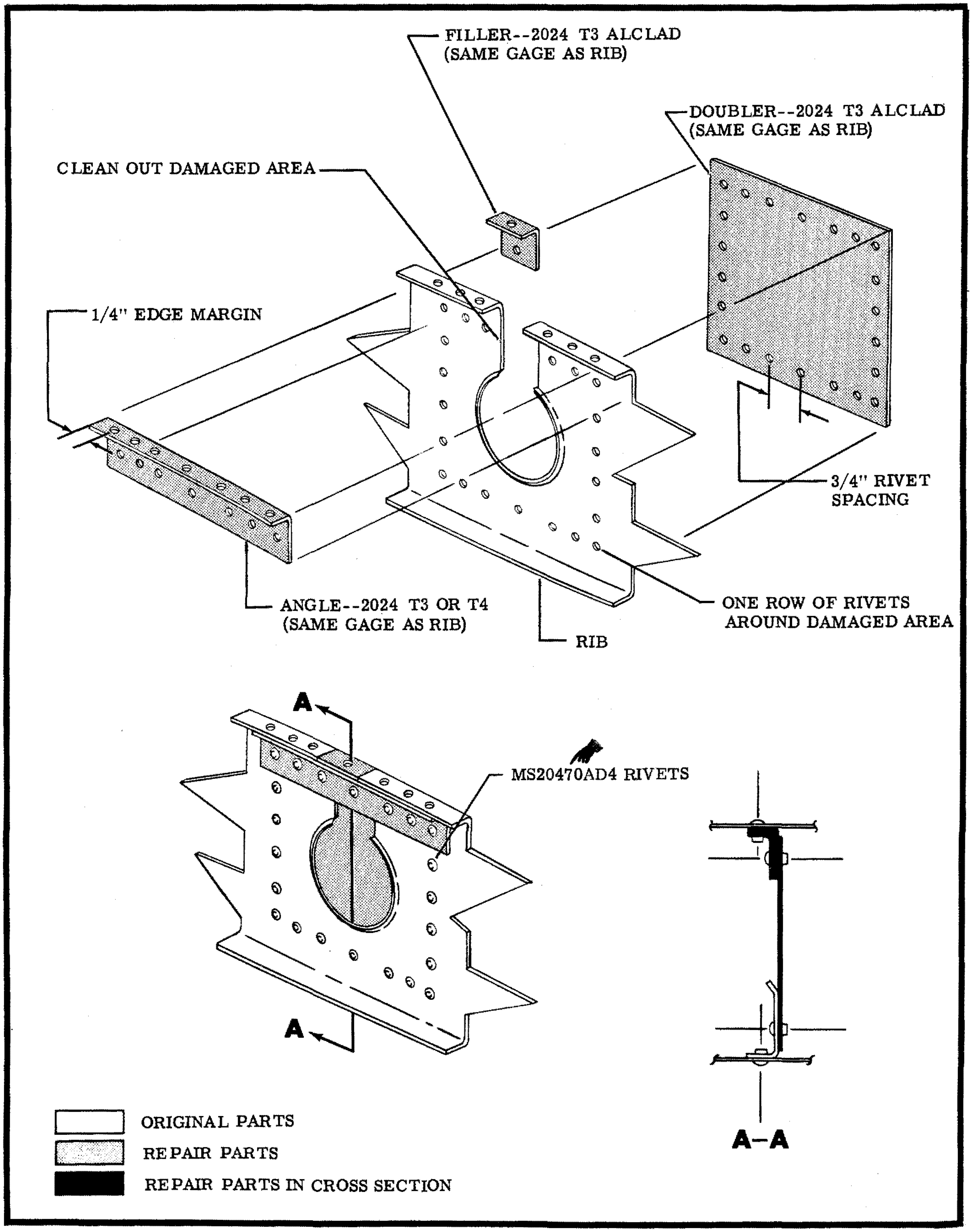



Figure 16-26. Typical Rib Flange Repair

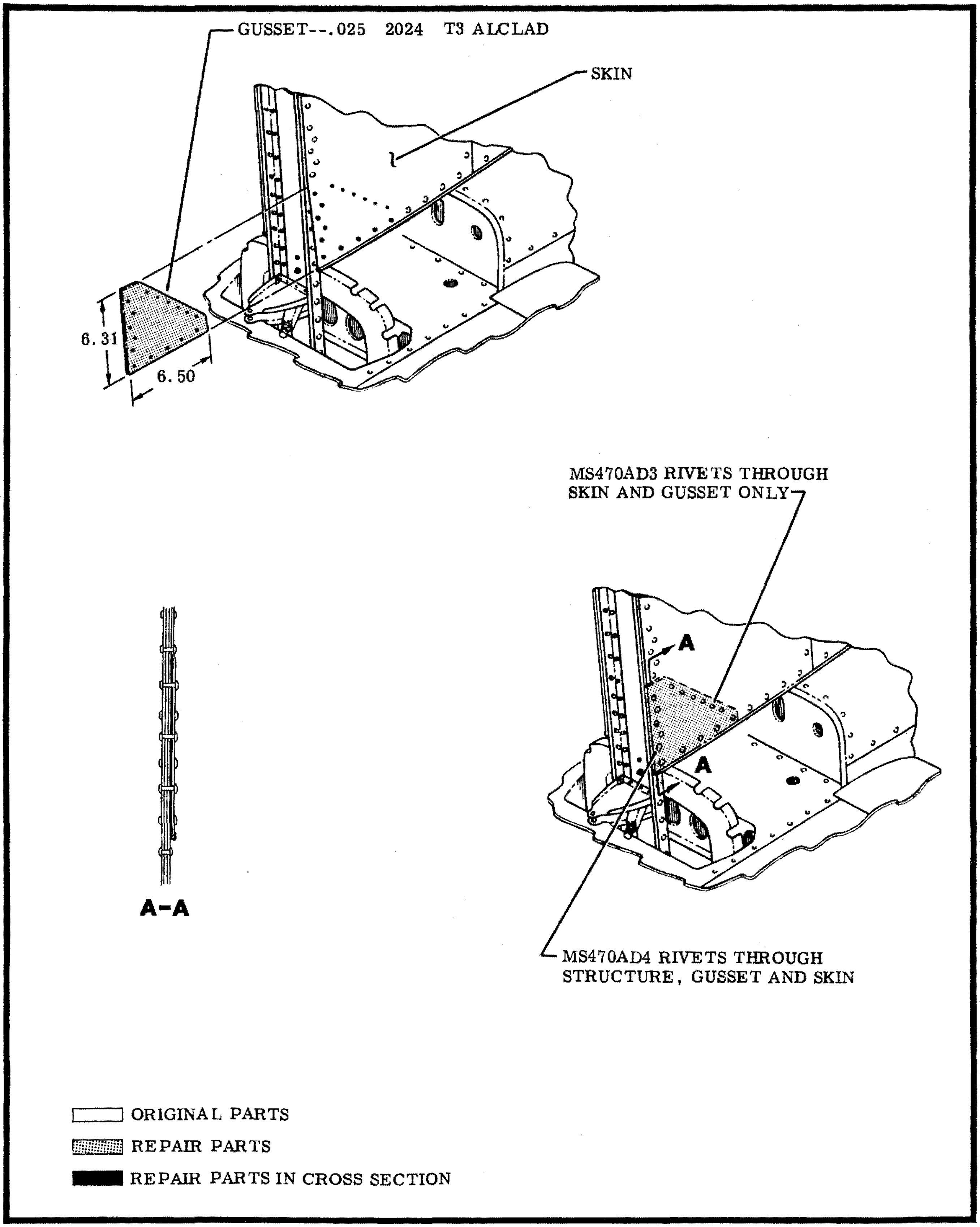



Figure 16-27. Typical Vertical Fin Repair

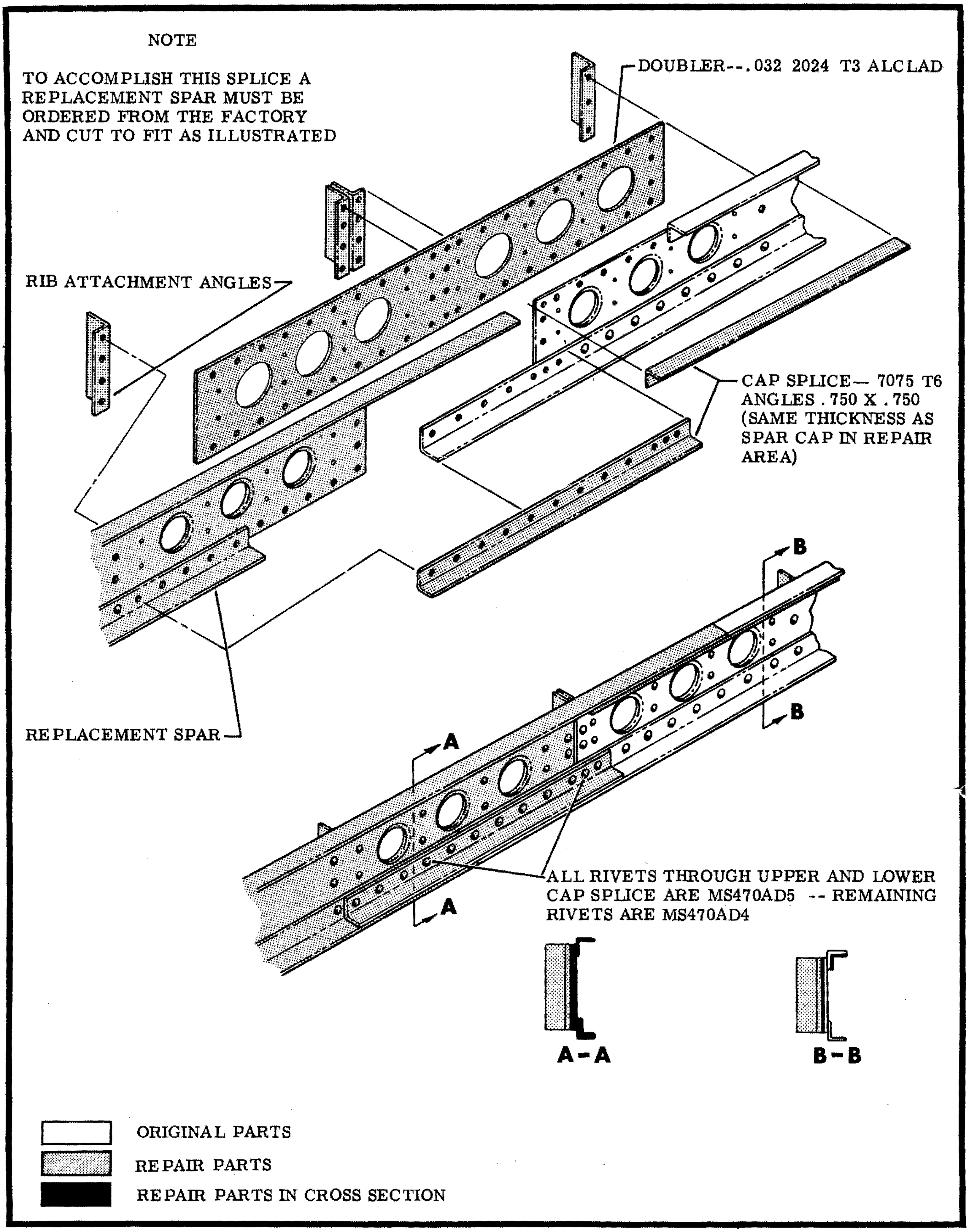



Figure 16-28. Typical Rear Spar Repair (Station 111.12 and outboard)

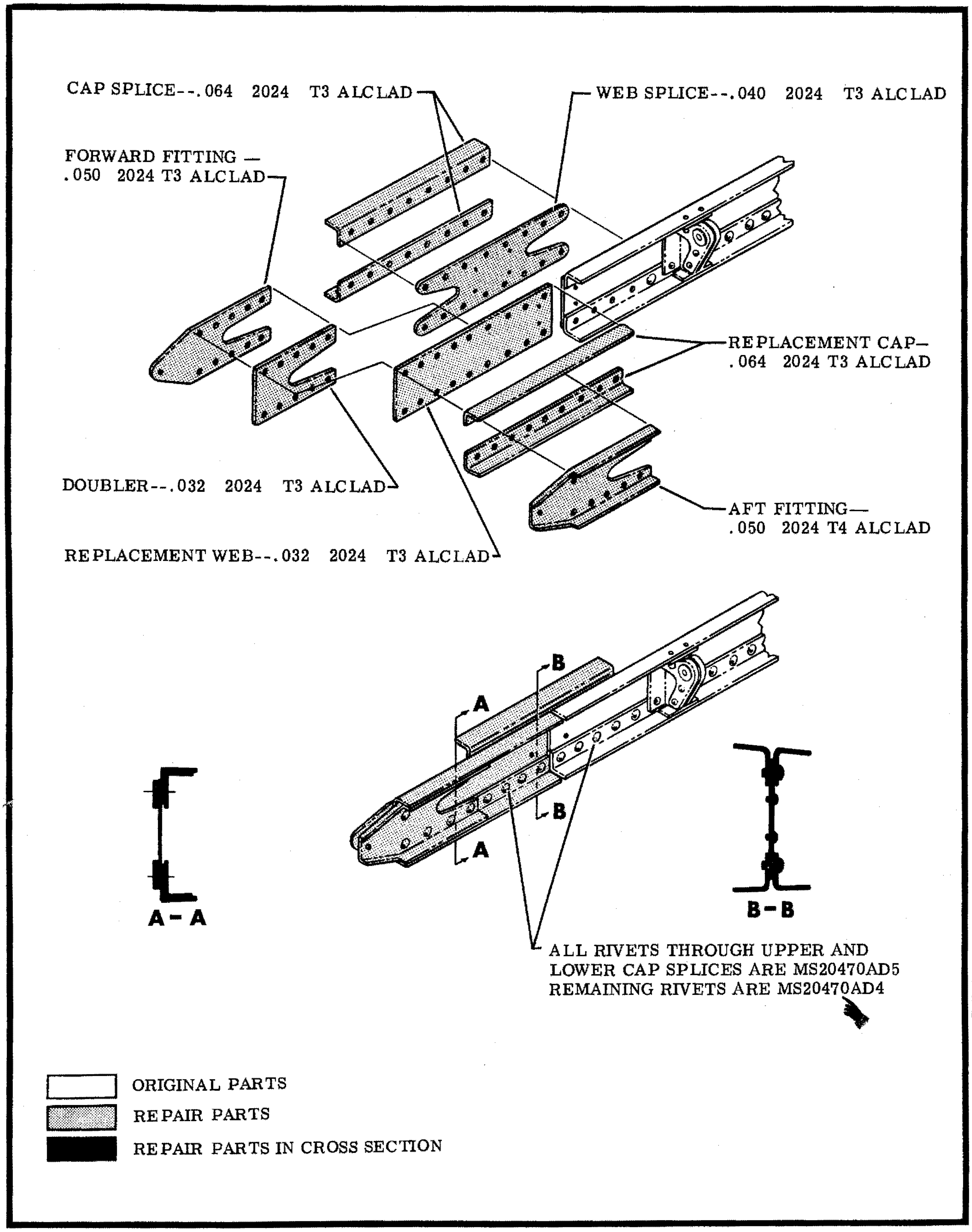



Figure 16-29. Typical Rear Spar Repair (Station 189.20 and outboard)

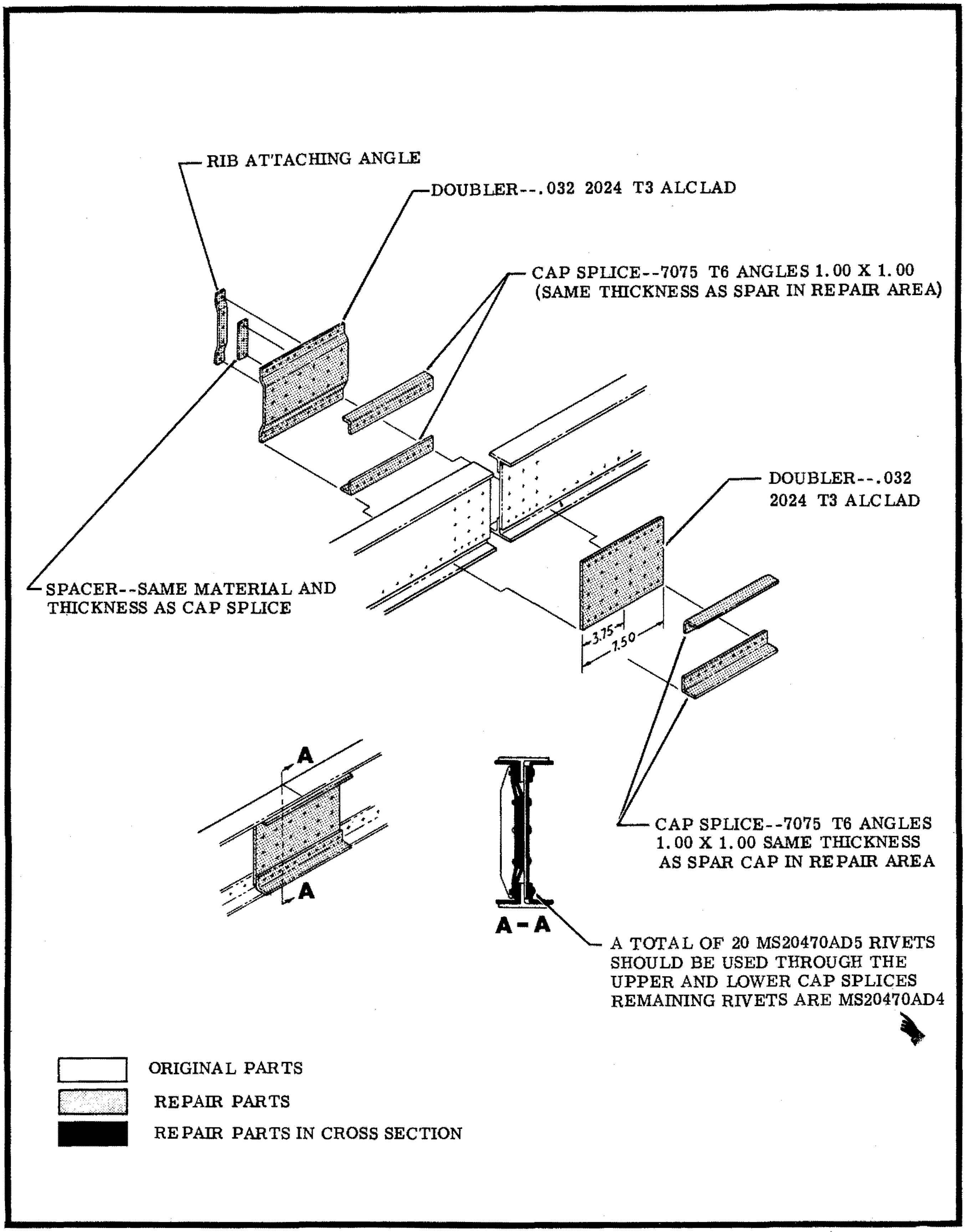



Figure 16-30. Typical Front Spar Repair (Station 168.75 and outboard)

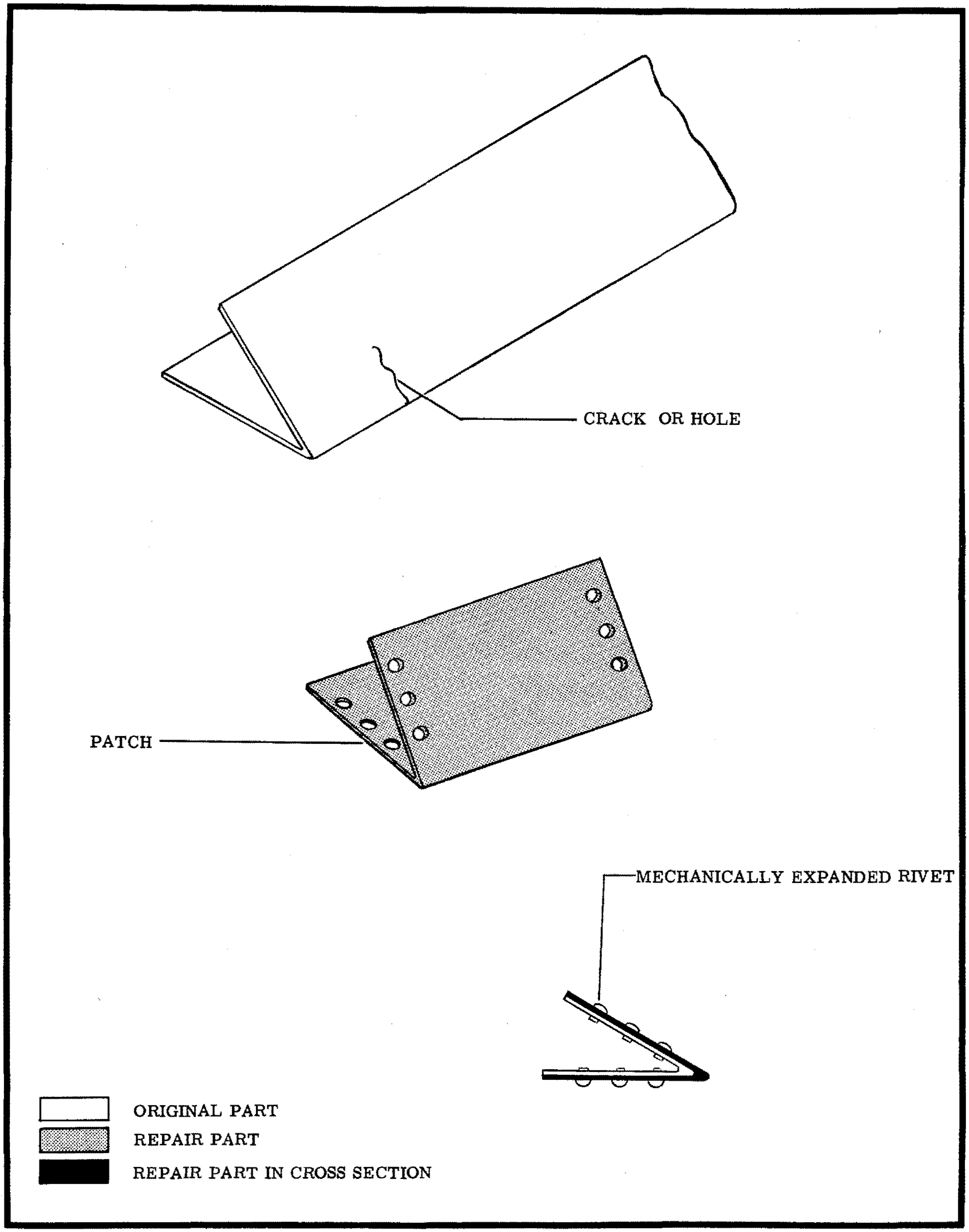



Figure 16-31. Typical Repair of Trailing Edge of a Control Surface

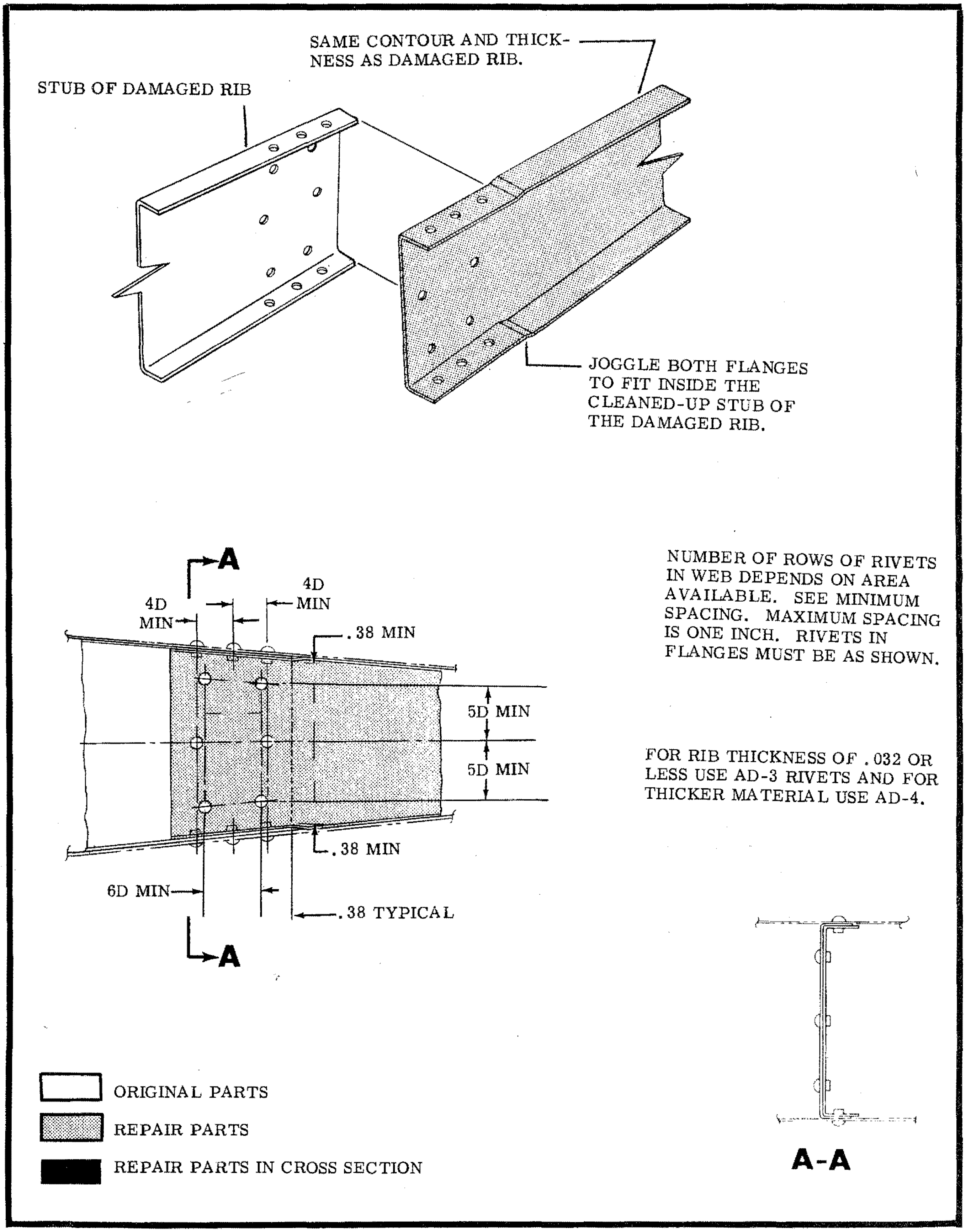



Figure 16-32. Typical Repair of Control Surface Rib

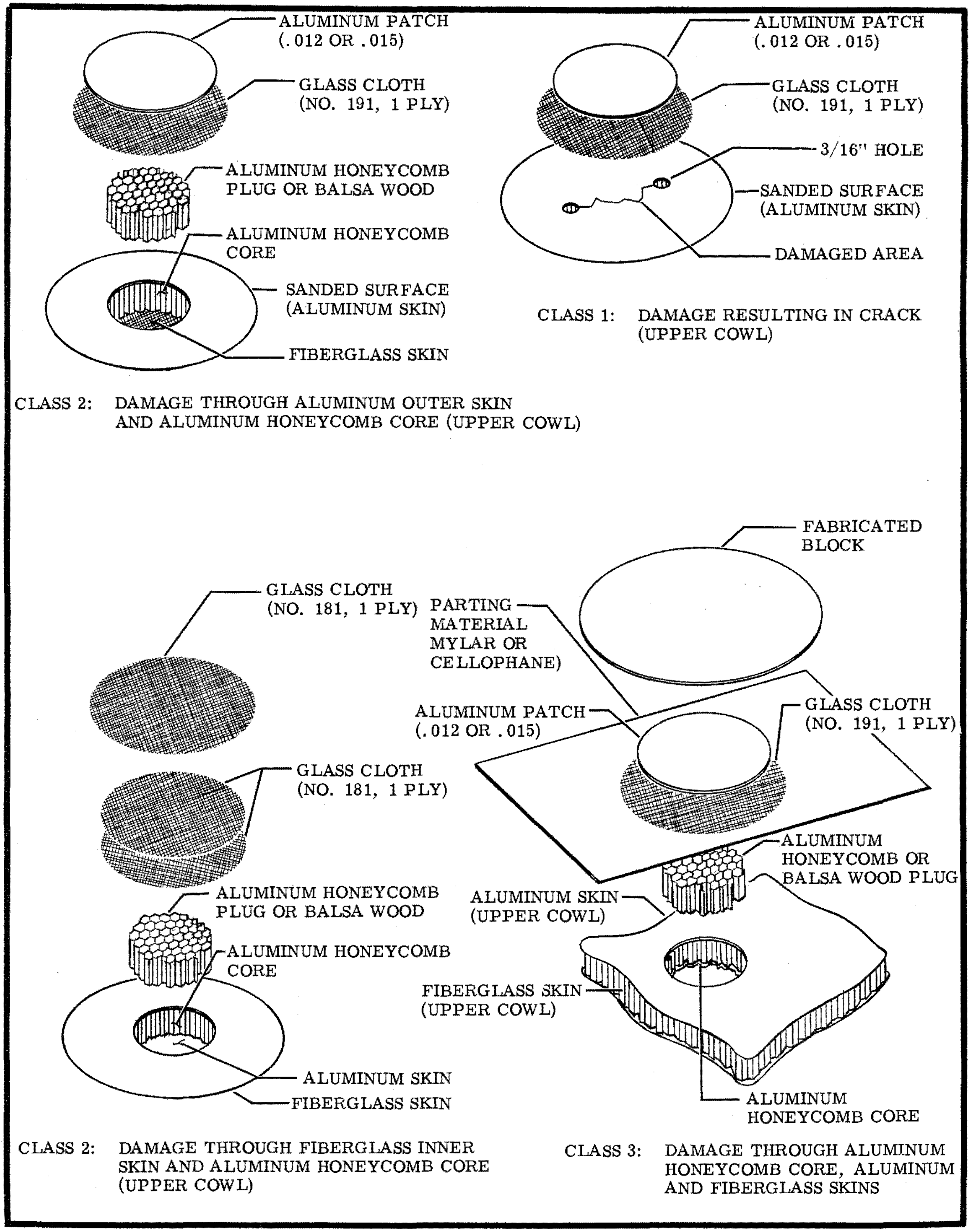



Figure 16-33. Repair of Upper and Side Engine Cowling

ALIGNMENT AND SYMMETRY CHECK. (See figure 16-34.)

Before making an alignment and symmetry check the aircraft should be defueled and leveled in accordance with Section 2. Figure 16-34 provides the measurements and shows the relative elevation points to be measured during the alignment symmetry check. Measurements are made with a steel tape projected between alignment points.

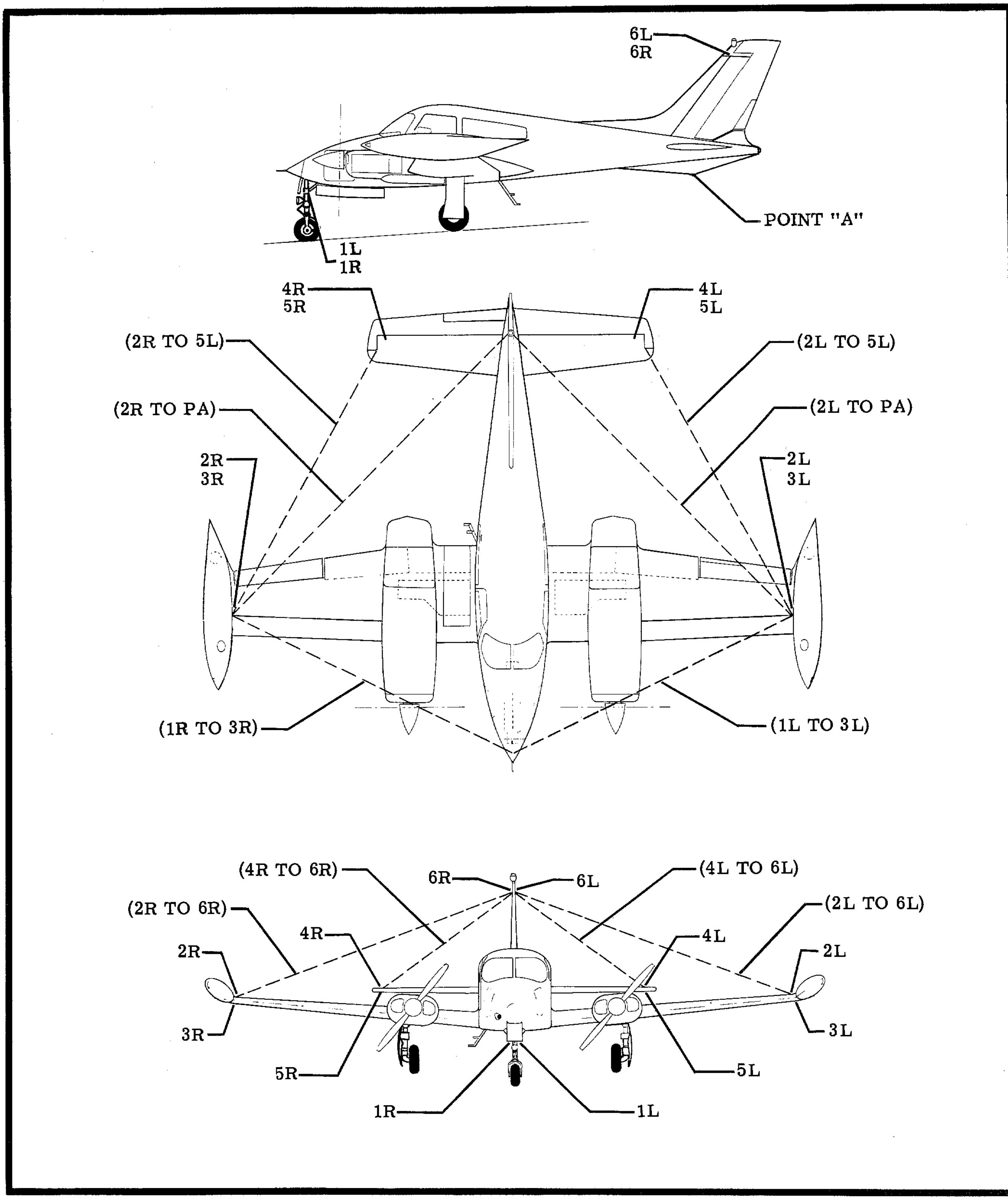
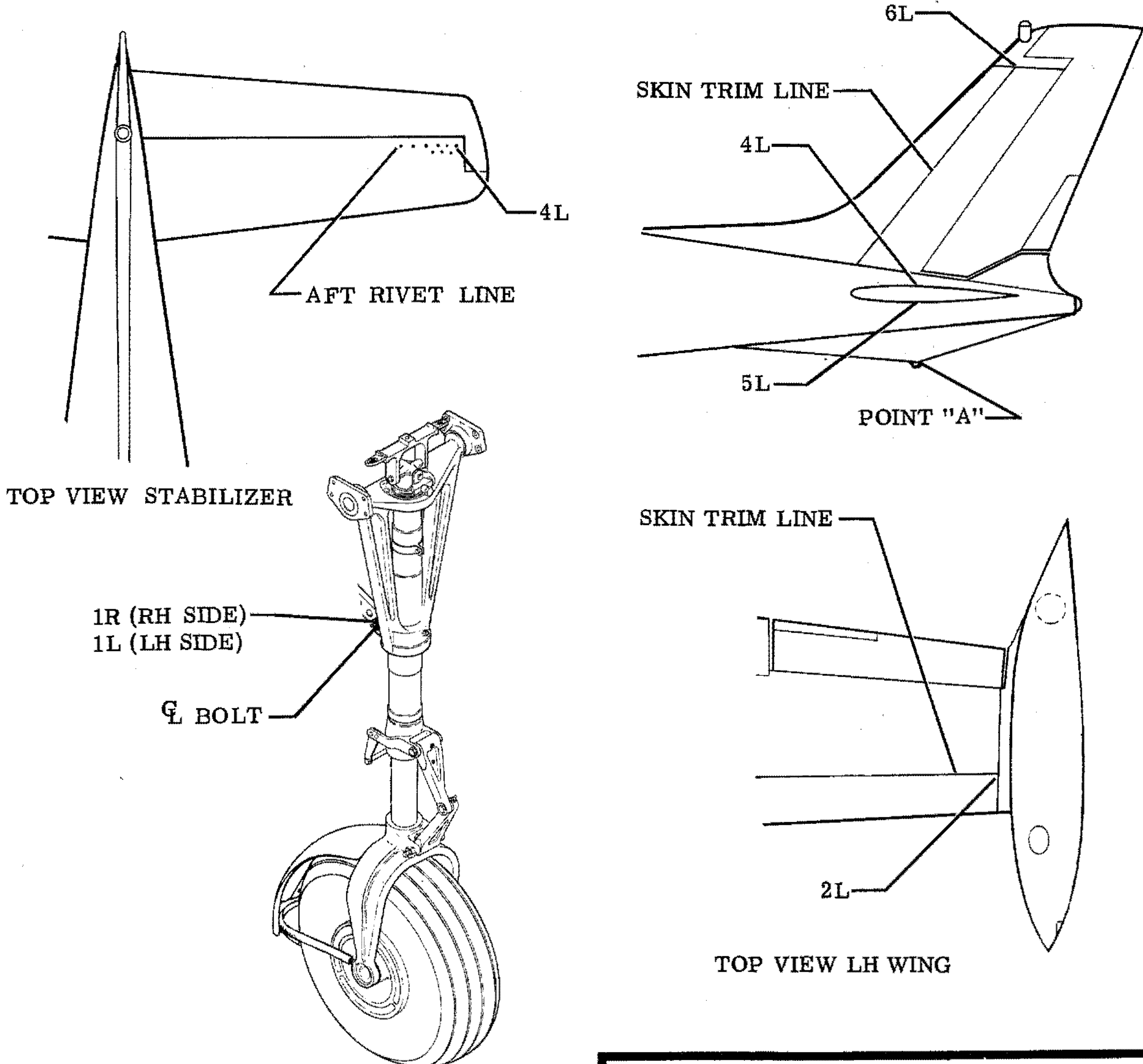



Figure 16-34. Alignment and Symmetry Check (Sheet 1 of 2)




| ALIGNMENT AND SYMMETRY CHECK | |
|---|---|
| POINTS | DISTANCE |
| 1L TO 3L<br>1R TO 3R | 18' 2.50" ±1.25" |
| 2L TO 5L<br>2R TO 5R | 18' 6.70" ±2.00" |
| 2L TO PA<br>2R TO PA | 23' 3.40" ±1.50" |
| 4L TO 6L<br>4R TO 6R | 9' 6.70" ±1.00" |
| 2L TO 6L<br>2R TO 6R | 24' 11.00" ±2.00" |

Details of Check Points:

A. Center line aft tail skid bolt.
1. Center line of bolt at gear strut and drag brace attachment. Center line head of bolt RH side and shank of bolt LH side.
2. Skin trim line and inboard edge of tip fairing at front spar.
3. Same as 2 except bottom of wing.
4. Center line aft rivet line of rear stabilizer at outboard edge of stabilizer.
5. Same as 4 except bottom of stabilizer.
6. Skin trim line at forward fin spar and lower edge of tip cap.

Figure 16-34. Alignment and Symmetry Check (Sheet 2 of 2)

CHECKING WING TWIST AND LOCATION OF THRUST LINE. (See Figure 16-35)

a. Remove wing in accordance with wing removal procedures and place wing on suitable supports beneath the root and tip ribs.

NOTE

The wing tip tank must be removed during check.

b. Locate wing datum plane as follows:

1. Locate a line at the root rib (wing station 28.40) which is 4.52 inches up from the lower surface of the front spar, and 4.00 inches up from the lower surface of the rear spar.

2. Locate a line at the tip rib (wing station 199.92) which is 1.44 inches up from the lower surface of the front spar, and .81 inches up from the lower surface of the rear spar.

3. These two lines locate the wing datum plane, and the three degrees of twist will be present if the lines are parallel.

c. Detail A, B and C, locate engine thrust line.

d. Install wing and wing tip tank in accordance with installation procedures.

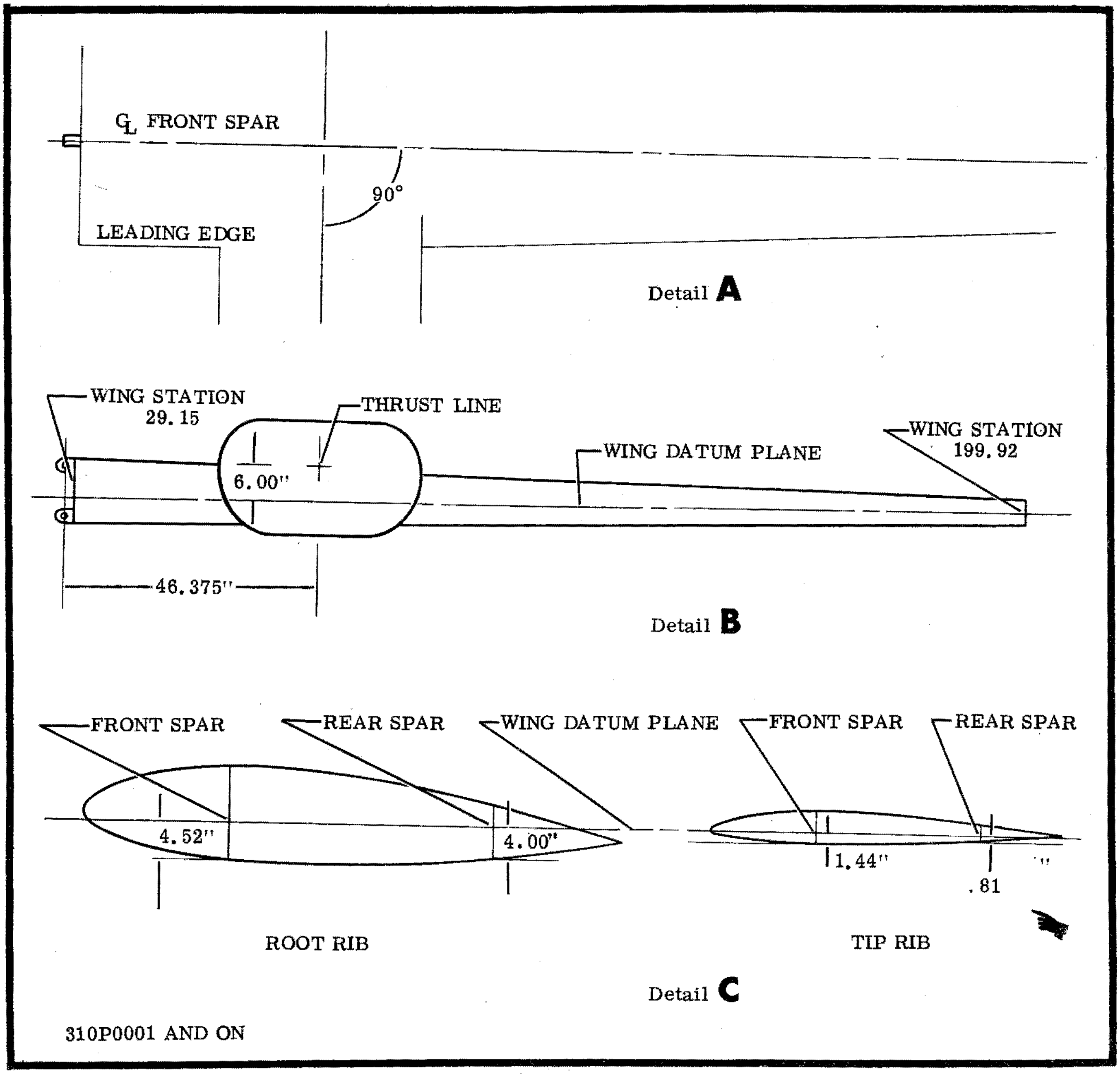



Figure 16-35. Wing Twist and Thrust Line Data (Sheet 1 of 2)

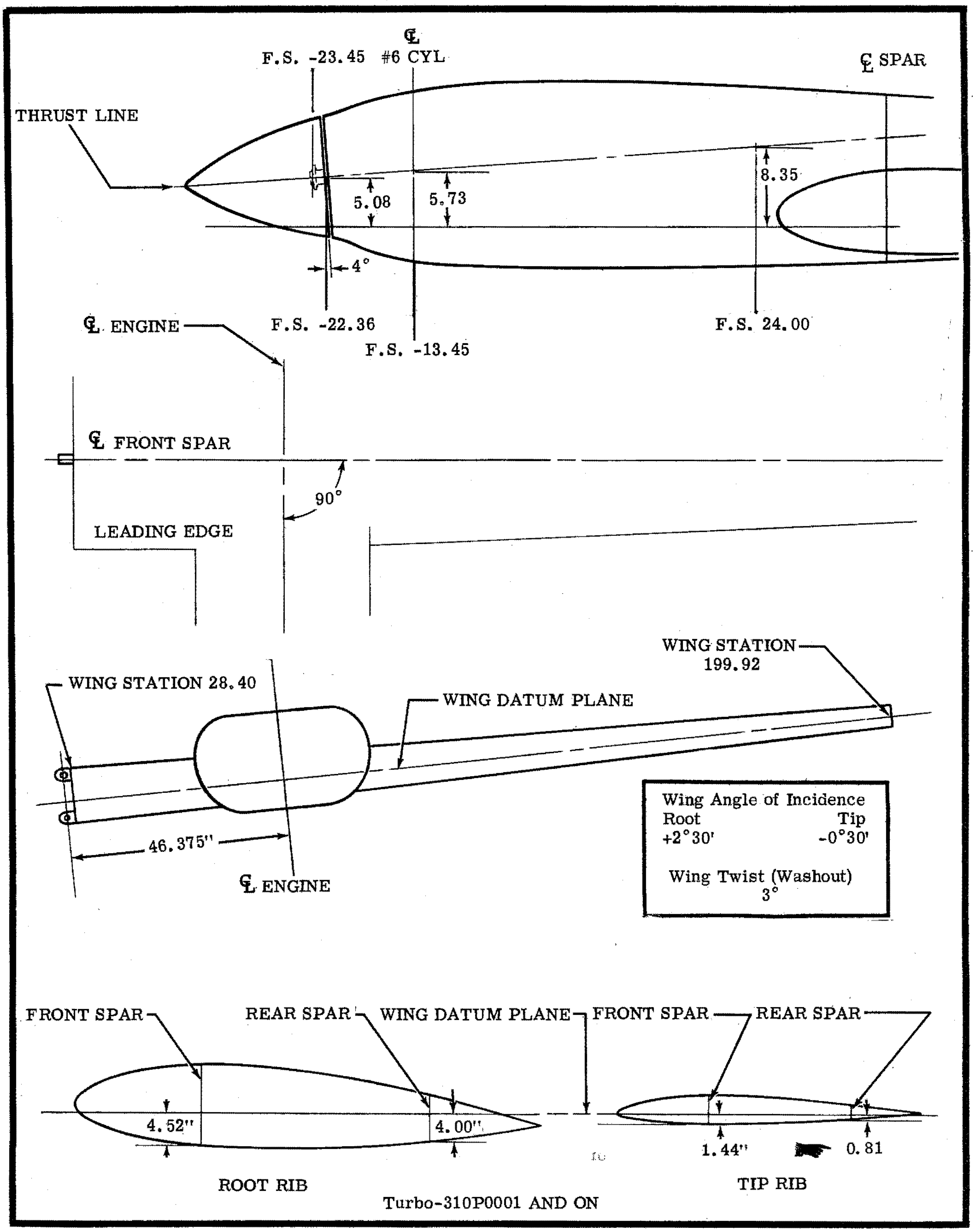



Figure 16-35. Wing Twist and Thrust Line Data (Sheet 2 of 2)

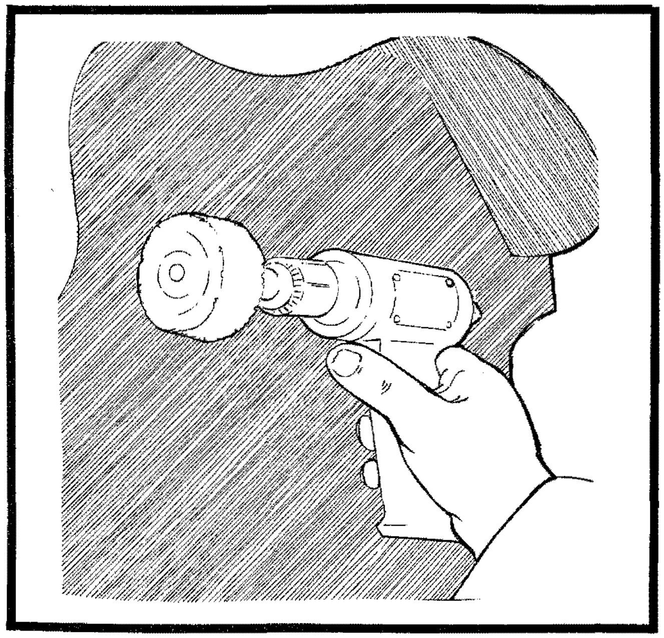


Figure 16-36. Buffing With Air Drill

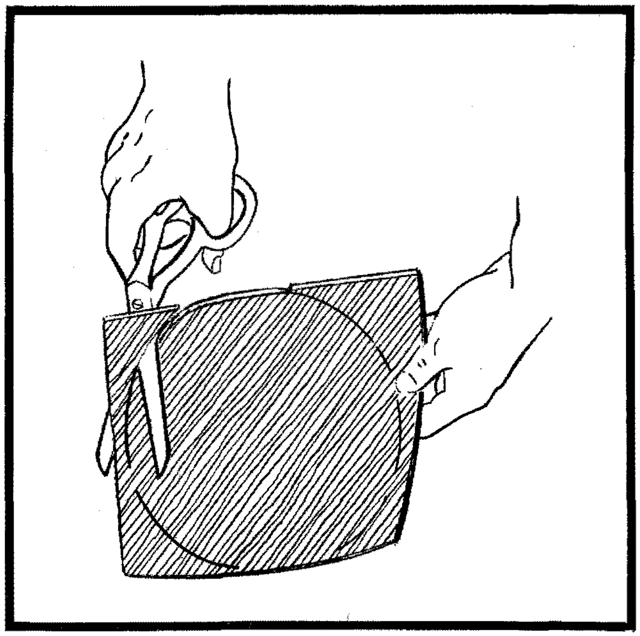


Figure 16-37. Fabrication of Patch

FUEL CELL REPAIR.

The fuel cells installed in the wings are a bladder-type, known as Construction BTC-39, manufactured by Goodyear Tire and Rubber Company. The following cold patch repair procedures are approved repair practices of Vithane BTC-39 fuel cells. These procedures apply only to BTC-39 fuel cells and deviations from these procedures should not be permitted.

CAUTION

Repair of the BTC-39 Vithane fuel cells is restricted to authorized personnel, and/or those certified or approved by factory trained schools.

a. The following equipment will be needed to make the cold patch repair:
1. Repair Kit, Dwg #2F1-3-35342, available from the Cessna Service Parts Center, consisting of the following:
   (a) 2331 Repair Cement (1/2 pint cans, 173 cc in each can).
   (b) 2328C Cross-linker (1 Oz. bottle, 28 cc in each bottle).
   (c) Methyl Ethyl Ketone (1 pint cans).
   (d) FT-160 Repair Fabric (sheet 12" x 12").
   (e) Cellophane (sheet 12" x 24").
   (f) Foam rubber, cloth back (sheet 12" x 12").
2. Paintbrush, 1-inch.
3. Roller, 1-inch diameter x 3/4" flat, or equivalent.

NOTE

The repair should be made on a well lighted table and natural contours maintained if possible. Prevent cell from contact with sharp edges, corners, dirty floors or other surfaces.

WARNING

Repair area must be well ventilated and no open flame or smoking in area of cell repair.

b. Repair Limitations.
1. FT-160 repair fabric is for repair of simple

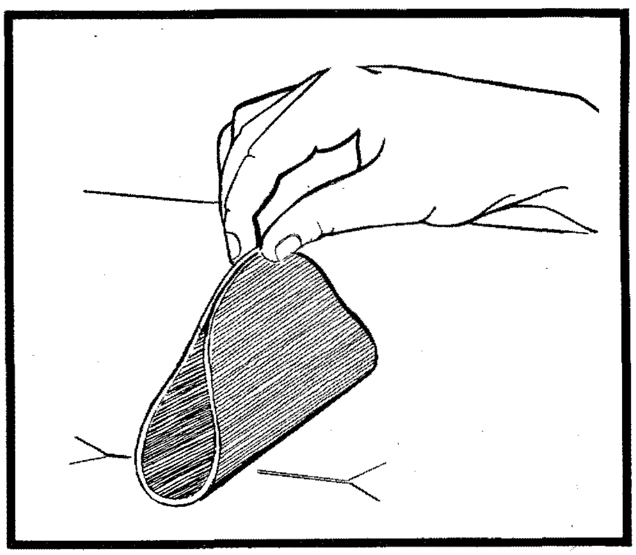


Figure 16-38. Centering Patch Over Injury

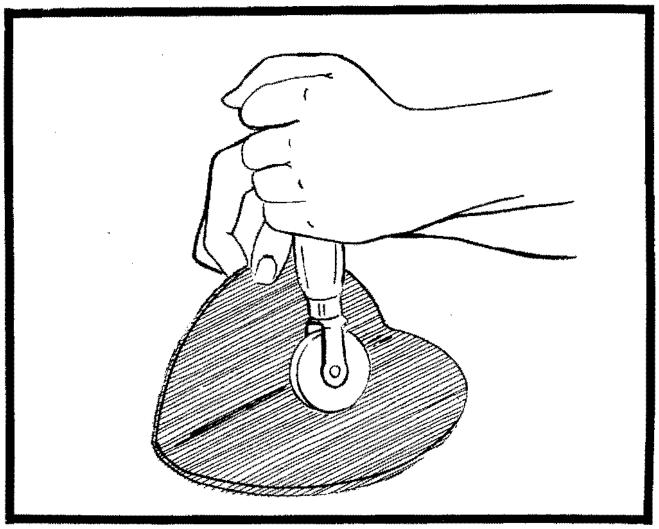


Figure 16-39. Use of Roller Stitcher

contours only. Patches referred to in these procedures are of this material.

2. Inside patches are to lap defect edges two-inches (2") in each direction.
3. Outside patches are to lap two and one-quarter inches (2-1/4") in each direction.
4. Outside patches are to be applied and cured prior to applying an inside patch.
5. Blisters between interliner and fabric larger than 1/4" in diameter, require an outside and an inside patch.
6. Separations between outer plies larger than one-inch (1") in diameter require an outside and inside patch. Holes and punctures require an outside and inside patch.
7. Slits and tears, up to three-inches (3") maximum length, require an outside and inside patch.
8. External abraided or scuffed areas, without fabric damage, require an outside patch only.
9. A loose lap may be trimmed, provided that a one-inch (1") effective bond remains.
10. Air cure repair patches are to remain clamped and undisturbed for 72 hours at room temperature of approximately 75°F.

c. Repair Patch.

1. Prepare exterior cell wall and exterior patch first. Cut repair patch from FT-160 material to size required (see figure 16-37) to insure proper lap over injury in all directions in accordance with Repair Limitations. Hole shears at an angle to produce a beveled edge (feather) on patch when cutting patch. Round corners of patch. Dull side or gum contact face of repair patch should be the largest surface after beveling.
2. Wash one square foot of cell wall surrounding injury, and repair patch contact side with a clean cloth soaked with Methyl Ethyl Ketone solvent.
3. Abraid cell wall surface about injury and contact side of patch with fine emery cloth or emery buffer wheel (see figure 16-36) to remove shine.
4. Repeat Methyl Ethyl Ketone washings two more times. A total of three washings each surface.
5. Tape a piece of cellophane inside cell over injury.
6. When all the above preparatory work has been completed, and all has been positioned for patch application on repair table, mix the 2331C cement (173 cc) with the cross-linker 2328C (28 cc), and stir mixture thoroughly.
7. Brush one even coat of mixed repair cement on the cell wall around injury and on the contact side of repair patch. Allow to dry for twenty minutes.
8. Repeat a second mixing of repair cement and brush on a second coat.

CAUTION

DO NOT USE FIRST CAN OF MIXED CEMENT FOR THIS COAT.

9. Allow cement to dry approximately ten minutes and then center patch over injury as shown in figure 16-38.
10. (See figure 16-39.) Lay repair patch by rolling down on surface from center of edge without trapping air. Hold the unrolled portion of repair patch off the cemented surface until roller contact insures an air-free union. At this time repair patch may be moved by hand on wet surface to improve lap. Do not lift repair patch, slide it.

CAUTION

Make sure cellophane inside cell over injury remains in place, because any cement will

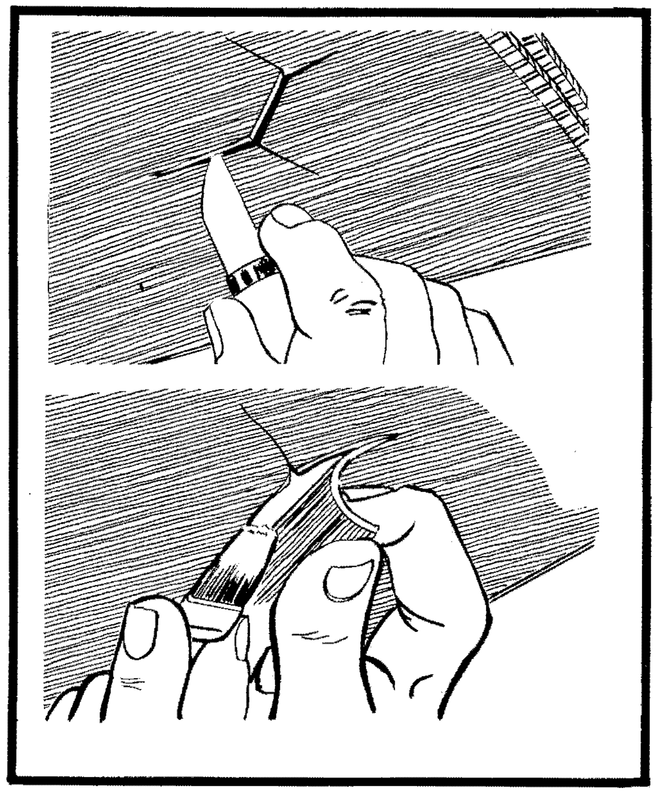


Figure 16-40. Repair of Blisters

stick cell walls together without it as a separator.

11. Cover one smooth surface each of two aluminum plates (plates must be larger than patch, with 1/4" fabric-backed airfoam, fabric side out. Tape airfoam in place. Foam must cover edges of plate for protection.
12. Fold cell adjacent to patch and place prepared plates, one over repair patch, and one on opposite side.
13. Secure the assembly with a "C" clamp, tighten by hand. Check cement flow to determine pressure.

CAUTION

Make sure that cell fold is not clamped between plates. This would cause a hard permanent crease. Also, be sure patch does not move when clamp is tightened.

14. Inside patch is applied in the same manner as the outside patch. See Repair Limitations paragraph for size of inside patch, after outside patch has been cured.

CAUTION

Success of applying both an outside and inside repair simultaneously is doubtful and not recommended.

d. Metal Fitting - Sealing Surfaces.
1. Rub off roughness of affected area with a fine file, emery cloth or emery wheel as shown in figure 16-36.
2. Clean metal surfaces using a clean cloth dipped in Methyl Ethyl Ketone. Moisten cleaned surface with clean cloth dipped in water. Apply Alodine 1200 solution, undiluted, to the affected area with a small nylon brush. Allow solution to dry until a light golden color appears. When coating has been formed, remove excess solution by wiping with a clean water-moistened cloth. Wipe dry.

WARNING

Do not allow solution to come in contact with hands, eyes or clothing.

e. Accessory Replacement.
1. Obtain cured repair accessory from manufacturer.
2. Mark location of old accessory and preserve marking for guide lines to locate new part.
3. Remove old accessory by gradually loosening an edge with a blunt probe-like instrument.
4. When an edge is loose, grasp accessory by loose edge with pliers and gently pull accessory off fuel cell wall. Be careful not to pull cell lap open while peeling accessory off. Pull from blind side of cell lap toward the exposed edge.
5. Buff the cell surface under accessory with emery cloth or emery buffer wheel (see figure 16-36) to smooth roughness and prepare for cement.

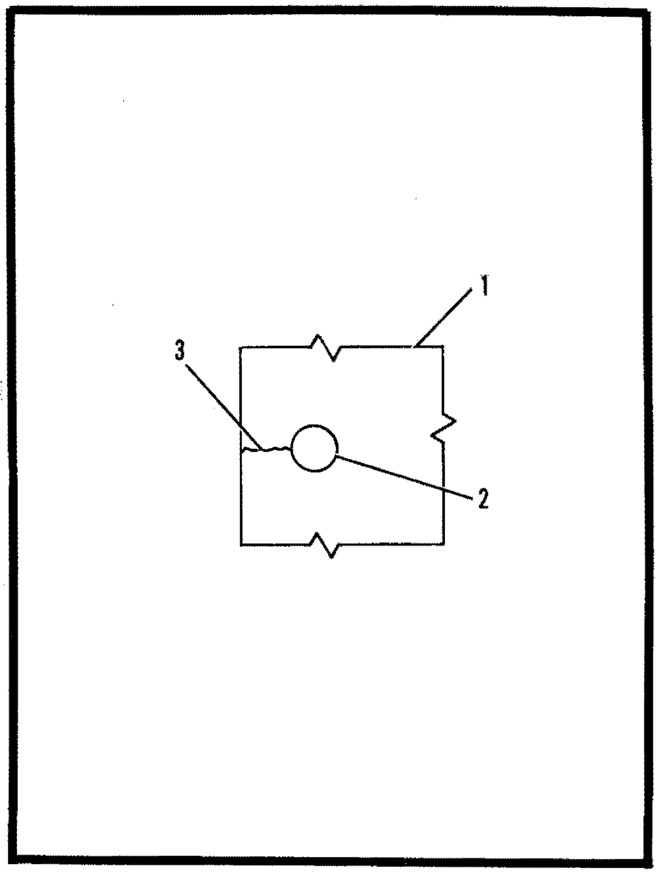
NOTE

Removal of accessory may leave an uneven cavity and surface. The surface under accessory should be buffed even and smooth.

6. Prepare replacement accessory by buffing and washing contact surface. Also wash cell surface (see Repair Patch paragraph).
7. Apply mixed 2331C repair cement to both surfaces.
8. Roll new accessory into place as with a repair patch and place suitable padded plates in position to insure adequate pressure when clamped. Use cellophane separator to prevent cement sticking in wrong places.
9. Cure as with repair patch procedure cure method.

f. Defects Repair.
1. Blisters (see figure 16-40).
   (a) Remove loose material by buffing or trimming. Apply an outside and inside patch.
2. Holes, punctures, cuts, tears and deep abraided areas.
   (a) Trim away any ragged material and apply an outside and inside repair patch.
3. Loose seams.
   (a) Buff loose edge and contact surface. Wash three times with MEK. Apply 2331C mixed cement, two coats, as with repair patch. Clamp and cure. Loose seams may be trimmed if minimum lap remains.
4. Loose fitting flange - Inside.
   (a) Buff edge of flange and contact surface under flange.
   (b) Apply 2331C mixed repair cement, cellophane padded plates and clamp.
   (c) Follow procedure as outlined for repair patch except patch itself.
5. Missing coat.
   (a) Buff surface.
   (b) Wash three times with MEK.
   (c) Apply 2331C mixed cement two coats as with repair patch.
   (d) Clamp and cure.
6. Looseness against metal.
   (a) Prepare metal as per metal fitting-sealing surface.
   (b) Apply 2331C mixed cement and cure.

g. Testing.
1. Attach test plates to all fittings.
2. Inflate the cell with air to a pressure of 1/4 PSI maximum.
3. Apply soap suds to patched area and check for leaks.




1. Ice Protection Panel 2. Rivet 3. Crack

Figure 16-41. Repairing Crack in Ice Protection Panel

REPAIR OF ICE PROTECTION PANELS.

The following repair procedures should be used for repairing the polycarbonate type ice protection panels. See figure 16-41 and 16-42 for types of cracks and method of repair. Figure 16-41 is an example of the cracks starting at a rivet hole propagating to the outer edge of the panel.

a. See figure 16-41, route the crack using router bit. Route material approximately 1/32 wide and 1/2 the depth of the panel thickness.

b. Clean routed crack with isopropyl or ethyl alchohol.

c. Fill crack with epibond 1331 in accordance with manufacturer's instructions.

NOTE

Epibond 1331 may be procurred from Furane Plastics, 4514 Brazil Street, Los Angeles, California 90039.

d. Figure 16-42 illustrates a combination of cracks which may occur. Cracks which do not extend to the outer edge of the ice protection panel shall be drilled through the ice protection panel with a #40 drill.

e. Fill the stop drilled hole with Epibond 1331 as per manufacturer's instructions.

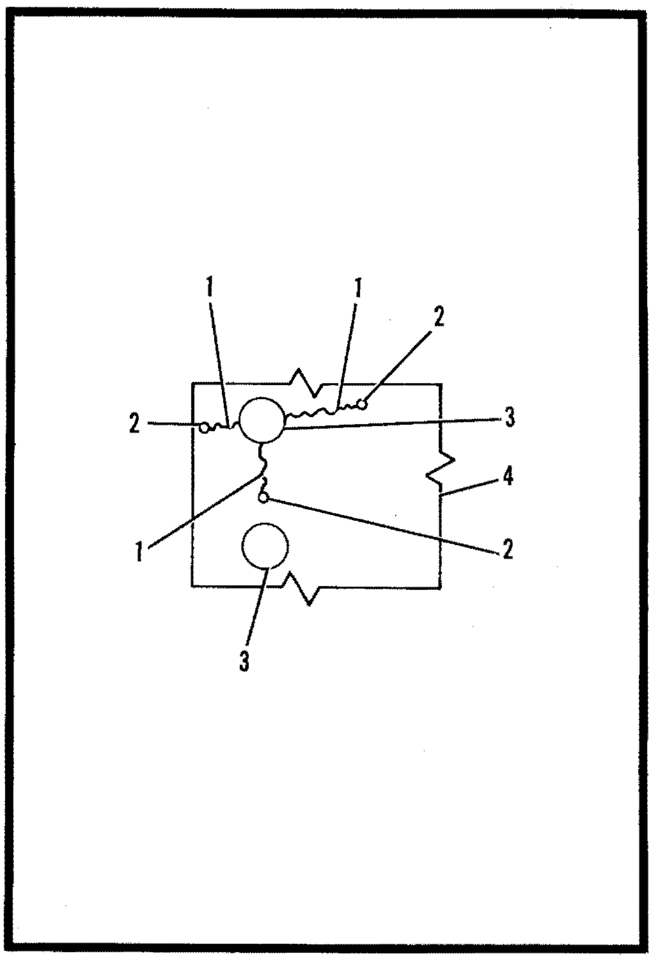
NOTE

Bondtite or Ditzler DX-666 may be used as an alternate for Epibond 1331.

f. Normal finishing, priming and painting shall be accomplished following the repair.

NOTE

Refer to section 2 for painting Polycarbonate.




1. Crack
2. Stop Drill
3. Rivet
4. Ice Protection Panel

Figure 16-42. Repairing Combination of Cracks in Ice Protection Panel

## RIVETS

### General.

The following rivets are commonly used in aircraft structures: standard solid shank, hi-shear, and blind. Rivets used in aircraft construction are most generally fabricated from aluminum alloys. In special cases monel, corrosion - resistant steel, mild steel, copper and iron rivets are used.

### Types.

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

b. Hi-shear rivets are special patented rivets having a high shear strength equivalent to that of the standard AN bolts. They are used in special cases in locations where high shear loads are present, such as spars, wings and heavy bulkhead ribs. The rivet consists of a cadmium plated pin of alloy steel and a collar of aluminum alloy. The installed rivet can be readily identified by the presence of the attached collar in place of the formed head on standard rivets.

c. Blind rivets are used where strength requirements permit when one side of the structure is inaccessible making it impossible or impractical to drive standard solid shank rivets.

### Substitution of Rivets.

a. Standard solid shank rivets.

1. In the replacement of rivets in installations which require the raised headrivets it is desirable to use whichever of the rivets that correspond to the type of rivet removed.
2. Countersunk head rivets are to be replaced by rivets of the same type and degree of countersink.
3. When rivet holes become enlarged, deformed, or otherwise damaged use the next larger size as replacement.
4. Replacements shall not be made with rivets of lower strength material unless they are larger than those removed.
5. In the absence of aluminum rivets, stainless, monel, or iron rivets may be used with generous application of zinc chromate primer only to permit necessary flights of the aircraft. The proper replacement shall be made as soon as facilities are available.

b. Hi-shear rivets. When hi-shear rivets are not available, replacement of sizes 3/16 or greater shall be made with bolts of equal or greater strength than the rivet being replaced and with self-locking nuts of the same diameter. The flush-type hi-shear rivet may be temporarily replaced by like diameter flush-type steel screws of equal or greater strength and self-locking nuts provided the threaded part of the screw does not extend into the material being fastened together. This latter procedure is temporary only and replacement with the hi-shear flush-type rivet must be accomplished as soon as they become available.

c. Blind rivets. Blind rivets have higher deflection rates in shear than standard solid rivets. For this reason, it is not advisable to replace any considerable number of solid rivets in a given joint by blind rivets inasmuch as this may result in over-stressing the remaining solid rivets. The following specific instructions apply.

1. The hollow blind rivet shall not be used.
2. The blind rivet used shall be of the same or greater strength and one size larger than the rivet it replaces, except that blind rivets may be replaced size for size.
3. In cases of dimpled assembly, the rivet holes shall be drilled after the sheets are dimpled.
4. When possible, the exposed end of each clipped plug shall be coated with a 10 per cent chromic acid solution or with zinc chromate primer.
5. Blind rivets shall not be used in hulls, floats, or tanks except in cases of absolute necessity.
6. If blind fasteners other than the blind rivets are encountered, it is recommended that replacements be made by either of these fasteners or by standard rivets.

### Diameters.

a. Rivet diameters range from 3/32 inch to 3/8 inch. The 1/8, 5/32 and 3/16 inch sizes are the most frequently used.

b. Since smaller rivets lack the proper structural qualities and larger rivets may dangerously reduce the splice or patch area, care must be exercised before substituting other than the specified sizes of rivet diameter.

### Lengths.

a. The proper length of rivet is an important part of the repair. Should too long a rivet be used, the formed head will be too large, or the rivet may bend or be forced between the sheets being riveted. Should too short a rivet be used the formed head will be too small or the riveted material will be damaged.

b. If proper length rivets are not available, longer rivets may be cut off to equal the proper length (not grip).

c. The rivet length is based on the grip.

### Removal of Solid Rivets. (See figure 16-43).

a. When it becomes necessary to replace rivets, great care should be taken in their removal so that the rivet hole will retain its original size and not acquire replacement with a larger size rivet.

b. To remove a rivet, file a flat on the manufactured head. It is always preferable to work on the original head rather than on the one that is bucked over, since the former will always be more symmetrical about the shank. Indent the flat surface with a counter-punch so the drill may be correctly centered. A drill slightly less in diameter than the rivet shank should be used to drill and weaken the head. Take care that the rivet shank does not turn with the drill and cause a tear. If the other end of the rivet is supported, the head may be sheared off with a sharp chisel. This cutting should always be done along the direction of the plate edge. If the shank is unduly tight after the removal of the head, the rivet should be drilled out completely. It may be forced out with a counterpunch of a smaller diameter than the rivet, provided the sheet is properly supported from the opposite side; however, there is greater danger of damaging the sheet and enlarging the hole when using this method. This procedure will also apply to flush rivets.

Riveting Installation. (See figure 16-44).

a. Riveting procedure. A large percentage of the riveting of aircraft structures is done on thin gage aluminum alloy and the work must be so accomplished that the material is not distorted by hammer blows or injured with riveting tools. All aircraft power riveting is done by upsetting or heading the rivets against a bucking bar instead of striking the shank with a hammer.

1. To prevent deforming of its head a rivet set must be selected to fit each type. The depth of this set must be such that it does not touch the material being riveted.
2. Parts which are to be heat-treated should be heat-treated before riveting since heat treating after this process causes warping. This is also necessary when assemblies are heated in a salt bath as the salt cannot be entirely washed out of the cracks.
3. Rivets of a diameter smaller than three thirty-seconds inch must not be used for any structural parts, control parts, wing covering, cowling, or similar sections of aircraft except where there are actual replacements.
4. Rivets through hollow tubes, which are loaded only in shear, should be hammered just enough to form a small head. No attempt should be made to form the standard round head as the amount of hammering required often causes the rivet to buckle inside the tube, with resultant injury to the member.
5. Aluminum alloy rivets must never be used in tension for structural, control, or other critical parts of aircraft. Whenever such an installation is required, bolts should be used.
6. The use of hollow rivets in joining highly stressed parts is not permitted. When rivets cannot be driven because of the inaccessibility of the end for bucking or driving, the next size self-plugging cherry rivet may be used.
7. The selection of the proper rivet and proper number of rivets is very important.
8. The rivets must be of the proper length for the total thickness of the pieces being riveted. Ordinarily, from 1-1/2 to 2 times the diameter of the rivet is about the right amount for the rivet shank to protrude through the material to form the head. For heavy material such as plates or fittings, from 2 to 2-1/2 diameters may be used.
9. The rivet should not be too loose in the hole as this condition will cause it to bend over while being headed, and the shank will not be sufficiently expanded to completely fill the hole. A drill from 0.002 to 0.004 inch larger than the rivet should be used for sheet and plate riveting.
10. Pieces should be held firmly together by clamps, screws or bolts while they are being drilled and riveted.
11. Where rivets are headed on the inside of the structure, the bucking bar is held against the end of the rivet shank. Care must be exercised doing this operation to prevent unseating the rivet by the application of too much pressure. For the first few blows the bucking bar should be held lightly against the rivet shank so that it will receive the impact of the blow through the rivet. The bucking bar must be held square with rivet to avoid turning it over.
12. Only a sufficient number of blows should be struct to properly upset a rivet. The blows must be as uniform as possible.

b. Sparing and diameter of rivets. There are no specific rules which are applicable to every case or type of riveting. There are, however, certain general rules which should be understood and followed.

1. The edge distance of rivets should not be less than two diameters of the rivet measured from the edge of the sheet or plate to the center of the rivet hole. (See figure 16-44).
2. The spacing between rivets, when in rows, depends upon several factors, principally the thickness of the sheet, the diameter of the rivets and the manner in which the sheet will be stressed. This spacing is seldom less than four diameters of the rivet, measured between the centers of the rivet holes. Rivets spaced four diameters apart are found in certain seams of monocoque and semimonocoque fuselages, webs of built-up spars, various plates or fittings, and floats or hulls.
3. Where there are two rows of rivets, they are usually staggered. The transverse pitch or distance between rows should be slightly less than the pitch of the rivets, 75% of the rivet pitch being the usual practice.
4. An average spacing or pitch of rivets in the cover or skin of most structures, except at highly stressed joints, will be from 6 to 12 diameters of the rivet.
5. The best practice in repair jobs is to make the pitch of the rivets equal to those in the original structure.

Loose or Working Rivets in Outboard Section of Wing.

Loose or working rivets attaching skin to upper and lower front spar cap may be repaired by adding MS20470AD4 or equivalent rivets midway between and in line with existing working rivets and four rivets beyond the last loose rivet, starting repair at the wing tip and working inboard. This repair is limited to the area between the nacelle and tip tank fitting.

NOTE

Care must be taken to avoid damage to fuel tanks and wiring.

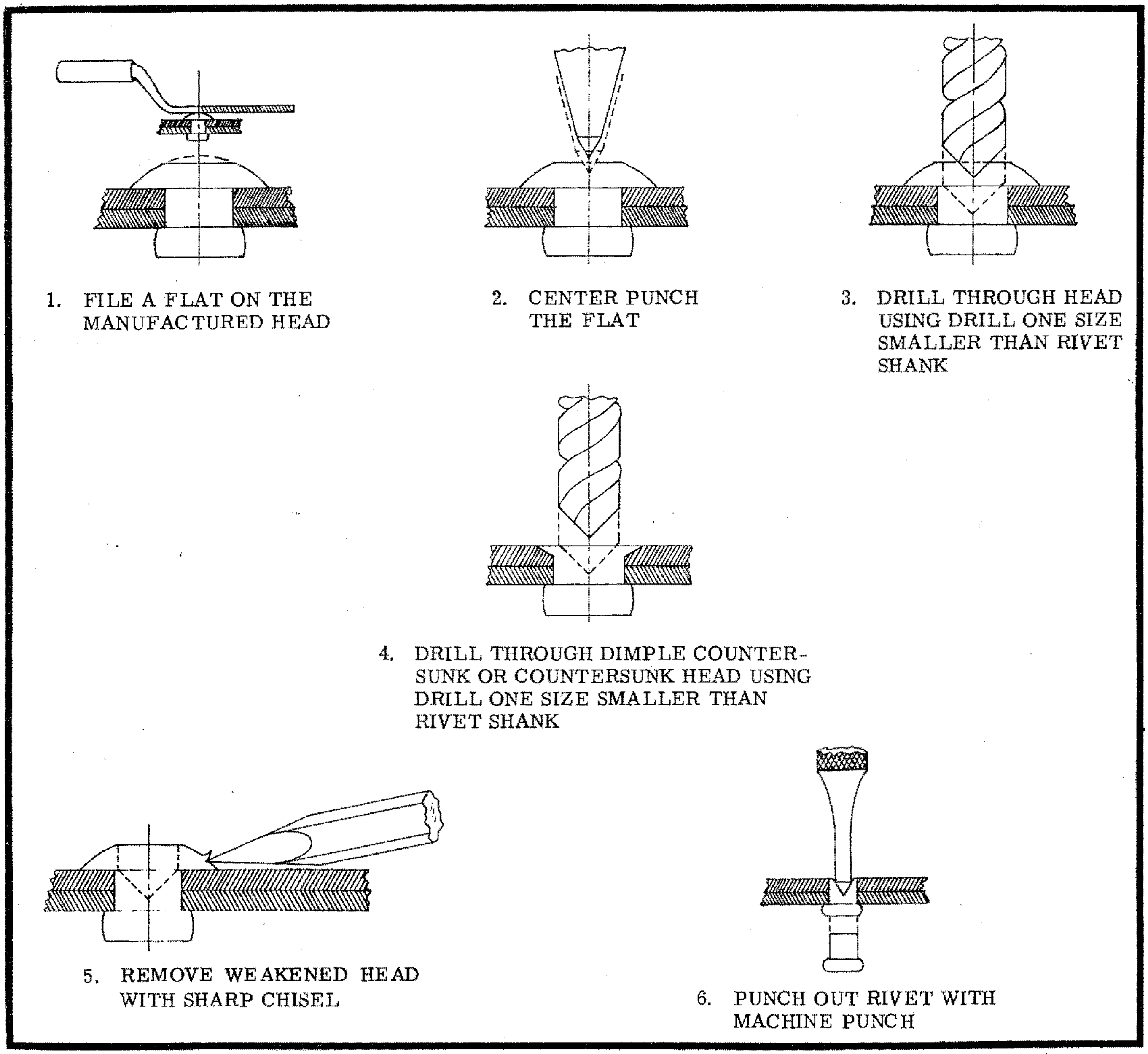



Figure 16-43. Removal of Rivets

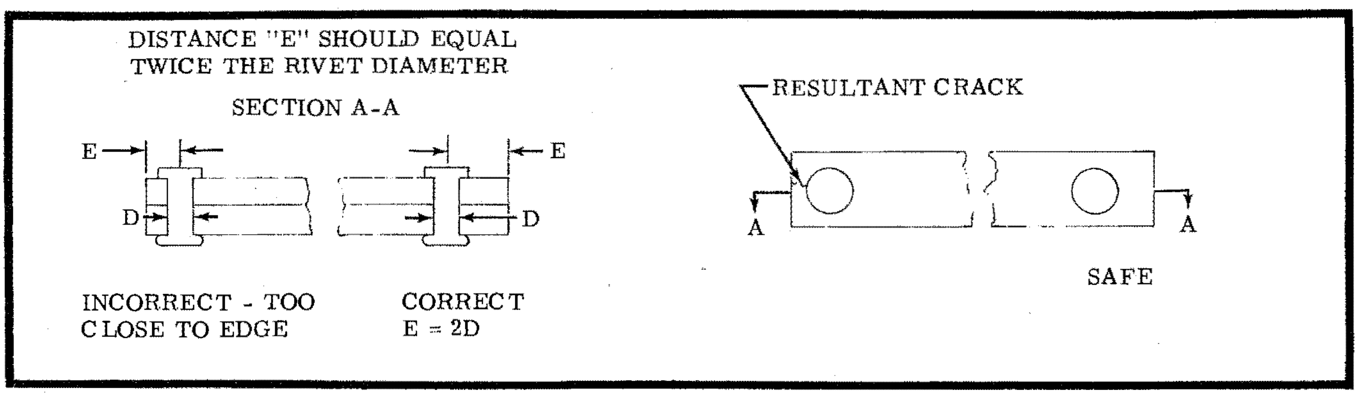



Figure 16-44. Rivet Edge Distance

Loose or Working Rivets.

a. Rivets which appear to be loose shall be checked by the use of a 0. 002 feeler gage by inserting the gage around the head of the rivet in question. If the feeler gage can be inserted to the shank of the rivet, it shall be classified as a loose rivet and shall be replaced.
b. If the feeler gage can be inserted approximately half-way to the shank for 30% of the circumference of the rivet head, it shall not be classified as a loose rivet.
c. The feeler gage shall be used to check the shear section between the riveted members, such as skin to spar or different sections of skins in a similar manner to that used around the rivet head.
d. If the skin around the brazier head or countersunk rivet can be moved by depressing the skin with finger pressure around the rivet, the rivet shall be replaced.
e. If rivets are found to turn by applying a rotating load to the head of the rivet, they should be replaced.
f. In areas where exterior paint has been applied to rivet heads, the paint may harden due to aging processes and show hairline cracks around the edge of the rivet heads. This should not be used as a basis for determining whether the rivet is loose or not. The hardened paint may crack at times and collect dirt or exhaust fumes which will appear as discoloration. It is not possible to detect loose rivets visually.
g. When replacing rivets, it is desirable to replace them with like size and type. In some instances, it will be necessary to go to the next size larger diameter. For general repair practices, the spacing between the centerlines of adjacent rivet holes shall be four diameters or greater. In areas where the spacing between rivets prohibits the use of the next size larger rivets, special repair instructions and procedures shall be utilized.

Resin Selectron 5003 - Comparable to MIL-R-7575 - Vendor: PPG Industries, Inc., One Gateway Center, Pittsburg, Penn., Attention: International Dept.
a. Alternate: Hetron 92 - Comparable to MIL-R-7575 - Vendor: Hooker Electrochemical Co., Niagra Falls, N. Y.

Luperson DDM - No MIL specification - Vendor: Lucidol Division, Wallace and Tiernan, Inc., Buffalo, N. Y.

Luperco ATC Paste - No MIL specification - Vendor: Lucidol Division, Wallace and Tiernan, Inc., Buffalo, N. Y.

Nuodex Cobalt - Federal Standard TT-D-642 - Manufacturer: Nuodex Canada Lmt., 34 Industrial St., Toronto 352, Ontario, Canada.