CIVIL AERONAUTICS MANUAL 18

Maintenance, Repair, and Alteration Of Certificated Aircraft, Engines, Propellers, and Instruments

August 1, 1949

U. S. DEPARTMENT OF COMMERCE
Charles Sawyer, Secretary

CIVIL AERONAUTICS ADMINISTRATION
D. W. Rentzel, Administrator

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INTRODUCTORY NOTE

Civil Aeronautics manuals are publications issued by the Civil Aeronautics Administration to implement and explain the Civil Air Regulations. The Civil Aeronautics manuals include the Civil Air Regulations and are the medium through which the public is apprised of CAA rules, policies, and interpretations.

CAA rules are supplementary regulations issued pursuant to authority expressly conferred on the Administrator in the Civil Air Regulations. Such rules are mandatory and must be complied with.

CAA policies provide detailed technical information on recommended methods of complying with the Civil Air Regulations. Such policies are for the guidance of the public and are not mandatory in nature.

CAA interpretations define or explain words and phrases of the Civil Air Regulations. Such interpretations are for the guidance of the public and will be followed by the Administration in determining compliance with the regulations.

This particular manual contains material interpreting and explaining the maintenance, repair, and alteration requirements specified in Civil Air Regulations, Part 18. The material pertaining to design, technique, and materials is arranged in 15 major sections, each of which contains supplementary material and general information on various subjects relating to CAR 18.20. For convenience the pertinent sections of CAR 18 are quoted in bold face type ahead of the manual material. Forms, tables, and figures relating to various sections of the manual are included in appendices A, B, and C, respectively.

The regulations quoted herein and the manual material are numbered according to a revised system which is used to facilitate publication of the contents in the Federal Register as required under the Administrative Procedures Act. For example, the CAR section identified as 18.20 is followed by 15 related CAM sections designated as 18.20–1, 18.20–2, etc. The numbering system is applied to paragraphs and subdivisions of paragraphs as follows:

18.20–3
- (a), (b), (c), etc.
  - (1), (2), (3), etc.
    - (i), (ii), (iii), (iv), etc.
      - *(a)*, *(b)*, *(c)*, etc.
        - *(1)*, *(2)*, *(3)*, etc.
          - *(i)*, *(ii)*, *(iii)*, *(iv)*, etc.

This revised edition of Manual 18 presents methods and techniques which, based on experience, have proved satisfactory in the repair and alteration of aircraft. Material relating to items of general interest also is presented. Inclusion of the latter type of information represents a departure from previous practice. It serves the purpose of acquainting mechanics and less experienced personnel with engineering aspects in fields with which they do not have frequent contact. It should be understood that any method or technique which can be shown to result in a degree of safety equal to one set forth in this manual will be acceptable to the Administrator of Civil Aeronautics. Any procedure or method shown to be inapplicable to a particular case may be suitably modified on request. In any event, the acceptance of any equivalent repair method or the modification of any procedure will become effective as of the date of approval, rather than the date of its incorporation in this manual.

This manual will be revised from time to time as equally acceptable methods or the need for additional explanations are brought to the attention of the Administrator of Civil Aeronautics.

Maintenance, Repair, and Alteration of Certificated Aircraft, Engines, Propellers, and Instruments

GENERAL

"CAR 18.1 *Definitions.* As used in this part: (a) 'Aircraft engine' means an aircraft engine approved by the Administrator. (b) 'Propeller' means a propeller approved by the Administrator. (c) 'Instrument' means an instrument installed, for other than purely experimental purposes, in a certificated aircraft. (d) 'Manufacturer' means: (1) the holder of the type certificate, or approval by the Administrator, for an aircraft, aircraft engine, or propeller, or of the current rights, under a licensing arrangement, to the benefits of such type certificate or approval, or (2) the manufacturer of a part or accessory of a certificated aircraft, or (3) the manufacturer of an instrument which is installed in a certificated aircraft: *Provided,* That such manufacturer shall have in his employ a properly certificated mechanic in direct charge of maintenance, repair, or alteration operations."

MAINTENANCE, REPAIRS, AND ALTERATIONS

"CAR 18.5 *Routine maintenance.* Routine maintenance is defined as simple or minor preservation operations including but not limited to the adjustment of rigging and clearances, and the replacement of small standard parts not involving complex assembly operations."

18.5–1 ROUTINE MAINTENANCE. *(CAA interpretations which apply to section 18.5.)*

The following are examples of routine maintenance operations:

(a) SERVICING OF AIRCRAFT.

Servicing on aircraft involving:

- Rigging.
- Adjustment of control surface and control system movements.
- Adjustment of landing gear retracting mechanisms, brakes, de-icing and electrical equipment, etc.

(b) SERVICING OF AIRCRAFT POWERPLANTS. Servicing and external adjustments on aircraft powerplants involving:

- Spark plugs.
- Cleaning ignition points.
- Valve tappets.
- Screens, etc.
- Hose connections in fuel, oil, and cooling systems, with identical parts made from same material, etc.

(c) SERVICING OF PROPELLERS. Servicing and adjustments of propellers involving:

- Smoothing out surface roughness of blades and polishing of propellers.
- Tightening of loose connections, etc.

(d) REPLACEMENT OF SMALL PARTS. Replacement of such small standard parts as:

- Bolts.
- Nuts.
- Pins.
- Bushings.
- Fair-leads.
- Pulleys.
- Turnbuckle terminals.
- Clamps.
- Hose connections in hydraulic systems, etc.
- Batteries, tires, tubes, windshield material, and other similar parts.

"CAR 18.6 *Repairs.* A repair is any operation other than routine maintenance

which is required to restore an aircraft, aircraft engine, propeller, or instrument to a condition for safe operation, including the mending or replacement of damaged or deteriorated parts.

"(a) *Minor repairs.* Minor repairs are elementary repair operations executed in accordance with standard practices and not within the definition of major repairs.

"(b) *Major repairs.* Major repairs are complex repair operations of vital importance to the airworthiness of an aircraft, including but not limited to: (1) Straightening, splicing, welding and similar operations when the strength of important structural members might be appreciably affected thereby. (2) Operations requiring complicated or unconventional techniques or equipment."

18.6–1 MINOR AIRCRAFT REPAIRS. *(CAA interpretations which apply to section 18.6 (a).)*

Repair or replacement work of the following types are considered minor aircraft repairs:

(a) NONSTRUCTURAL MEMBERS. Repairs to nonstructural members which may affect the airworthiness of an aircraft, such as:

Cowlings.
Turtlebacks.
Wing and control surface fairings.
Electrical installations.
Windshields.

(b) TANKS. Patching and repairing of leaks in nonintegral fuel, oil, water ballast, hydraulic, and de-icer fluid tanks.

(c) RIBS, LEADING AND TRAILING EDGES, TIP STRIPS. The repair of: Not more than two adjacent wing or control surface ribs of a conventional type (wood or metal); the leading edge of wing and control surfaces between two adjacent wing or control surface ribs; the trailing edge of wings, control surfaces, and flaps; the wing and control surface tip strips.

(d) CONTROL CABLES. Replacement of control cables.

(e) FABRIC COVERING.

Patching of fabric.

Replacement of the fabric covering of surfaces involving an area not greater than required to repair two adjacent ribs.

(f) METAL OR PLYWOOD STRESSED COVERING. The patching of holes in metal or plywood stressed covering not to exceed 3 inches in any direction when ribs, stringers, bulkheads, and reinforcements are not directly affected.

(g) REPLACEMENT OF COMPLETE COMPONENTS OR UNITS. Replacement of complete components or units such as listed below with parts supplied by the original manufacturer or manufactured in accordance with approved drawings.

Wings.[1]
Replaceable wing tips.[1]
Control surfaces.[1]
Wing or control surface bracing (struts or wires).
Sea wings.
Floats.
Wheels.
Skis.
Landing gears.
Tail wheel assemblies.
Engine mounts (prefabricated and bolted on, not to be welded on).
Fuel and oil system accessories.
Powerplant cowling.
Intake or exhaust systems.
Fuel and oil tanks.
Powerplant controls.
Propeller controls.
Instruments and safety belts.

18.6–2 AIRCRAFT ENGINE MINOR REPAIRS. *(CAA interpretations which apply to section 18.6 (a).)*

Overhaul work of the following types are considered aircraft engine minor repairs:

(a) TOP OVERHAULS. Top overhauls of engines of less than 200 horsepower involving the following:

Removal of cylinders.

[1] After completing replacement of wings, wing tips, and control surfaces, the airplane must be test-flown prior to returning it to service.

Grinding valves and removing carbon.
Fitting new rings.
Adjustment of valve gear or replacement of parts in valve mechanism except the rotating parts in the crankcase.

(b) COMPLETE OVERHAULS. Complete overhauls of:

engines with an M.C. (maximum continuous) rating of less than 200 horsepower, or
engines with an M.C. rating more than 200 horsepower which have neither an integral supercharger or integral propeller reduction gearing.

18.6-3 PROPELLER MINOR REPAIRS. *(CAA interpretations which apply to section 18.6 (a).)*

Repairs to wood or metal propellers, propeller hubs, and propeller governors when made in accordance with the repair practices and methods prescribed in this manual, or in accordance with "Administrator of Civil Aeronautics Approved" recommendations of the propeller manufacturers, are considered minor propeller repairs.

18.6-4 INSTRUMENT MINOR REPAIRS. *(CAA interpretations which apply to section 18.6 (a).)*

Cleaning, adjustment, or replacement of parts in instruments such as are listed below are considered minor instrument repairs:

Air-speed indicators.
Altimeters.
Rate of climb indicators.
Compasses.
Turn and bank indicators.
Engine tachometers.
Temperature and pressure gages.
Fuel flow meters.
Electrical and gyroscopic instruments.

(a) REPLACEMENT OF REMOVABLE INSTRUMENTS. Replacement of removable instruments may be considered an aircraft minor repair in accordance with CAM 18.6-1.

18.6-5 AIRCRAFT MAJOR REPAIRS.[2] *(CAA interpretations which apply to section 18.6 (b).)*

Repairs of structural components and elements of components such as outlined below, usually found necessary as the result of an accident or of unforeseen conditions requiring repair or reinforcement, are considered to be major aircraft repairs, unless classified as minor aircraft repairs in CAM 18.6-1.

(a) RIBS AND LEADING EDGES. Repair of three or more wing or control surface ribs or the leading edge of wings and control surfaces between such ribs.

(b) FUEL TANKS. Rebuilding including rebottoming of standard type or integral fuel tanks.

(c) FABRIC COVERING. Repair of fabric covering involving a greater area than required to repair two adjacent ribs; replacement of fabric on fabric-covered components such as wings, fuselages, control surfaces, etc.

(d) METAL OR PLYWOOD STRESSED COVERING. The repair of damaged areas in metal or plywood stressed covering exceeding three inches in any direction; the repair of portions of skin sheets by making additional seams; splicing of skin sheets.

(e) STRUCTURAL REWORK INVOLVING REPAIR OF HIGHLY STRESSED MEMBERS. All repairs involving the replacing, strengthening, reinforcing, and splicing of highly stressed members such as:

Spars.
Spar flanges.
Members of truss-type beams.
Thin sheet webs of beams.
Keel and chine members of boat hulls or floats.
Corrugated sheet compression members which act as flange material of wings or tail surfaces.
Wing main ribs and compression members.
Wing or tail surface brace struts.

[2] See CAR 18.11 *Provision for Approval of Major Repairs and Major Alterations.*

Fuselage longerons.
Members of the side truss, horizontal truss, or bulkheads.
Main seat support braces and brackets.
Landing gear brace struts.
Axles.
Wheels.
Skis, and ski pedestals.
Parts of the control system such as control columns, pedals, shafts, or horns.

18.6–6 AIRCRAFT ENGINE MAJOR REPAIRS. *(CAA interpretations which apply to section 18.6 (b).)*

Work of the following types is considered aircraft engine major repair:

(a) OVERHAUL. Top and complete overhaul of engines with an M.C. (maximum continuous) rating of more than 200 horsepower which have either an integral supercharger and/or propeller reduction gearing.

(b) SPECIAL REPAIRS. Repairs to engine parts by welding or any other means. (See CAM 18.20 for technique and practices.)

18.6–7 PROPELLER MAJOR REPAIRS. *(CAA interpretations which apply to section 18.6 (b).)*

Repairs of the following types are considered major propeller repairs:

(a) REPAIRS TO WOOD OR METAL PROPELLERS, PROPELLER HUBS, AND PROPELLER GOVERNORS. Repairs to wood or metal propellers, propeller hubs, and propeller governors not made in accordance with repair practices and methods as prescribed in this manual or in accordance with "Administrator of Civil Aeronautics Approved" recommendations of the propeller manufacturer.

(b) REPAIR OF STEEL BLADES. The repair or straightening of steel blades.

(c) REPAIR OF STEEL HUBS. The repair or machining of steel hubs.

(d) SHORTENING OF BLADES. The shortening of blades below the minimum diameter indicated in the Civil Aeronautics Administration Propeller Specifications so as to materially affect the performance of the propeller.

18.6–8 INSTRUMENT MAJOR REPAIRS. *(CAA interpretations which apply to section 18.6 (b).)*

No attempt is made at present to differentiate between minor and major instrument repairs, and therefore all instrument repairs may be considered minor repairs.

"CAR 18.7 *Alterations.* An alteration is any appreciable change in the design of an aircraft, aircraft engine, propeller, or instrument.

"(a) *Minor alteration.* A minor alteration is: (1) An alteration having no appreciable effect on the weight, balance, structural strength, power-plant operation, flight characteristics, or other characteristics affecting the airworthiness of an aircraft; or (2) An alteration for which specific plans and instructions have been approved by the Administrator and which can be executed by means of elementary operations.

"(b) *Major alterations.* Major alterations are all alterations not within the definition of minor alterations."

18.7–1 AIRCRAFT MINOR ALTERATIONS. *(CAA interpretations which apply to section 18.7 (a).)*

Changes such as listed below are considered minor aircraft alterations. (See CAM 18.20–15 for details of equipment changes.)

(a) OPTIONAL EQUIPMENT ON AIRCRAFT SPECIFICATION.—The installation or removal of specific items of optional equipment listed in the Aircraft Specification[3] when made in accordance with the manufacturer's instructions.[4]

(b) EQUIPMENT OF EQUAL OR LESS WEIGHT. The installation or removal of equipment of equal or less weight and in the same location and in the same manner as that listed as optional equipment in the Aircraft Specification.

(c) SIMPLE MODIFICATIONS. Changes of relatively minor nature, such as the addition of reinforcements or fittings, which are easily installed and the installation

[3] Copies of Aircraft Specifications may be obtained from CAA Aviation Information, Washington, D. C.
[4] The cumulative weight changes of all such alterations since the last Civil Aeronautics Administration inspection should not exceed 2 percent of the certificated weight empty.

of which does not require appreciable rework of the aircraft structure, when made in compliance with "Airworthiness Directives" which supplement the Aircraft Specifications[5] issued by the Civil Aeronautics Administration, or in accordance with "Administrator of Civil Aeronautics Approved" alteration instructions of the manufacturer of the aircraft.[6]

(d) CHANGES TO IMPROVE SERVICE LIFE. Changes of a minor nature on structural and nonstructural elements for the purpose of improving the service life or reducing maintenance costs, provided the cumulative weight changes of such alterations do not exceed 2 percent of the certificated weight empty since the last Civil Aeronautics Administration inspection, and provided the empty weight center of gravity location is not changed.

18.7–2 AIRCRAFT ENGINE MINOR ALTERATIONS. ***(CAA interpretations which apply to section 18.7 (a).)***

The alteration or conversion of an aircraft engine by replacement or addition of parts, in compliance with "Airworthiness Directives" listed on the engine specifications issued by the Civil Aeronautics Administration, or in accordance with "Administrator of Civil Aeronautics Approved" alteration instructions of the manufacturer of the aircraft engine, is considered a minor aircraft engine alteration.

18.7–3 PROPELLER MINOR ALTERATIONS. ***(CAA interpretations which apply to section 18.7 (a).)***

Changes on propellers, hubs, or propeller governors made in compliance with "Airworthiness Directives" which supplement the Propeller Specifications issued by the Civil Aeronautics Administration, or in accordance with "Administrator of Civil Aeronautics Approved" alteration instructions of the propeller manufacturer, are considered minor propeller alterations.

[5] Copies of Aircraft Specifications may be obtained from CAA Aviation Information, Washington, D. C.

[6] Proper reference to the pertinent request by the Civil Aeronautics Administration or alteration bulletin of the manufacturer should always be made in the repair records of the repair agency and in the aircraft logbook for the benefit of the Aviation Safety Agent conducting the follow-up inspection.

18.7–4 INSTRUMENT MINOR ALTERATIONS. ***(CAA interpretations which apply to section 18.7 (a).)***

Design changes made in accordance with the "Administrator of Civil Aeronautics Approved" recommendations of the manufacturer of the instruments are considered minor instrument alterations.

18.7–5 AIRCRAFT MAJOR ALTERATIONS. ***(CAA interpretations which apply to section 18.7 (b).)***

Changes of the following type when not listed in the Aircraft Specifications are considered major aircraft alterations because they are likely to affect the airworthiness of an aircraft to a great degree. (See CAM 18.20–15 for details of equipment changes).

(a) INSTALLATION OR REMOVAL OF EQUIPMENT.[7] The installation or removal of equipment of any type in any location other than outlined in CAM 18.7–1.

(b) DESIGN CHANGES. Basic design changes on any component such as:

Wings.
Tail surfaces.
Fuselage.
Landing gear, etc.
Elements of components (spars, ribs, shock absorbers, bracing, cowlings, turtlebacks, fairings, balance weights, etc.) of an aircraft.

(c) CHANGES IN FUEL, OIL, COOLING, HEATING, DE-ICING, AND ELECTRICAL SYSTEMS. Changes over the original design in the fuel, oil, cooling, heating, de-icing, electrical, and exhaust systems of an aircraft. This includes conversions of

[7] Alterations are considered to fall under the special privilege provisions contained in CAR 18.11 for certificated repair stations and manufacturers without having to be executed in accordance with any other manual or further specification approved by the Administrator, if they are carried out so that: (a) The cumulative weight change of all such alterations since the last Civil Aeronautics Administration inspection does not exceed 2 percent of the certificated weight empty; (b) the cumulative effect upon the empty weight center of gravity of all such alterations since the last Civil Aeronautics Administration inspection does not exceed ½ of 1 percent of the M.A.C.; (c) the method of attachment to the structure is such as to distribute the weight uniformly over as large a portion of the structure as practical; (d) the installation is such as to have no adverse effect upon the original airworthiness. (For instance, equipment increasing the fire hazard or otherwise interfering with the safe operation of an aircraft should not be considered satisfactory for installation.)

any sort for the purpose of using fuel of rating or grade other than that called for in original approval.

(d) INSTALLATION OF DIFFERENT TYPE ENGINE, PROPELLER, OR FEATHERING SYSTEM. The installation of a type of engine or propeller different from that approved for the airplane or the installation of propeller feathering or deicing systems.

18.7-6 AIRCRAFT ENGINE MAJOR ALTERATION. *(CAA interpretations which apply to section 18.7 (b).)*

Changes of the following type are considered major aircraft engine alterations:

(a) ENGINE CONVERSION. Conversion of an aircraft engine from one approved type to another, involving any changes in compression ratio, propeller gear, or impeller gear ratios, which requires extensive rework and testing of the engine, though all the work may be accomplished by using factory parts and carried out in accordance with the aircraft engine manufacturer's "Administrator of Civil Aeronautics Approved" alteration instructions.

(b) MODIFICATION WITH UNAPPROVED PARTS. When aircraft engine structural parts are replaced with parts other than those supplied by the original manufacturer or specifically approved by the Civil Aeronautics Arministration[8] for the purpose of obtaining approval of the new parts.

18.7-7 PROPELLER MAJOR ALTERATIONS. *(CAA interpretations which apply to section 18.7 (b).)*

Changes in blade design, hub, or propeller governor not authorized in the Propeller Specifications issued by the Civil Aeronautics Administation[9] are considered major propeller alterations.

18.7-8 INSTRUMENT MAJOR ALTERATIONS. *(CAA interpretations which apply to section 18.7(b).)*

Changes in design not made in accordance with the "Administrator of Civil Aeronautics Approved" recommendations of the instrument manufacturer are considered major instrument alterations.

RULES AND PROCEDURES FOR MAINTENANCE, REPAIRS, AND ALTERATIONS

"CAR 18.10 *Agencies authorized to perform maintenance, repair, and alteration operations.* Maintenance, repair, and alteration operations shall be performed only by:

(a) A certificated mechanic having the proper rating or a person working under the direct supervision of such mechanic; or (b) a certificated repair station having the proper rating; or (c) the manufacturer of the aircraft or part of the aircraft to be repaired: *Provided*, That all instrument repairs and alterations and propeller major repairs and major alterations shall be performed only by a certificated repair station having the proper rating or by the instrument or propeller manufacturer."

18.10-1 REPAIR AGENCIES. *(CAA interpretations which apply to section 18.10.)*

Agencies which repair or alter an aircraft, aircraft engine, propeller, or appliance in accordance with the classifications set forth in subsections of CAM 18.6 and CAM 18.7 are classified as:

Certificated repair stations.
Manufacturers.
Certificated mechanics.

(a) CERTIFICATED REPAIR STATION. A certificated repair station means a repair station certificated in accordance with the provisions of Civil Air Regulations (CAR 52).

(b) MANUFACTURER. For the purpose of this manual a manufacturer means (1) the holder of the type certificate, or approval by the Administrator, for an aircraft, aircraft engine, or propeller, or of the current rights, under a licensing arrangement, to the benefits of such type certificate or approval; or (2) the manufacturer of a part or accessory of a certificated aircraft; or

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[8] Changes as above require extensive proof tests as specified in Part 13 of the Civil Air Regulations and CAM 13. Also see CAM 18.20-13 (a) (4).

[9] Changes such as outlined above usually involve proof testing of the propeller or governor in accordance with CAR 14.

(3) the manufacturer of an instrument which is installed in a certificated aircraft: *Provided,* That such manufacturer shall have in his employ a properly certificated mechanic[10] in direct charge of maintenance, repair, or alteration operations.

(c) CERTIFICATED MECHANIC. A certificated mechanic means a mechanic certificated in accordance with the provisions of the Civil Air Regulations (CAR 24).

18.10–2 AGENCIES AUTHORIZED TO PERFORM MAINTENANCE, MINOR REPAIRS, AND MINOR ALTERATIONS. ***(CAA interpretations which apply to section 18.10.)***

(a) AIRCRAFT MAINTENANCE, MINOR REPAIRS, AND MINOR ALTERATIONS. Maintenance, minor repairs, or minor alterations of a certificated aircraft must be made by one of the following:

A certificated repair station holding the appropriate rating.
The manufacturer of the aircraft.
A certificated mechanic holding an aircraft mechanic rating.
A person under the direct supervision of a certificated mechanic holding an aircraft mechanic rating.

(b) AIRCRAFT ENGINE MAINTENANCE, MINOR REPAIRS, AND MINOR ALTERATIONS. Maintenance, minor repairs, or minor alterations of a certificated aircraft engine must be made by one of the following:

A certificated repair station holding the appropriate rating.
The manufacturer of the aircraft engine.
A certificated mechanic holding an aircraft engine mechanic rating.
A person under the direct supervision of a certificated mechanic holding an aircraft engine mechanic rating.

(c) AIRCRAFT PROPELLER MAINTENANCE, MINOR REPAIRS, AND MINOR ALTERATIONS. Maintenance, minor repairs, or minor alterations of a certificated propeller must be made by one of the following:

A certificated repair station holding the appropriate rating.
The manufacturer of the propeller.
A certificated mechanic holding an aircraft engine mechanic rating.
A person under the direct supervision of a certificated mechanic holding an aircraft engine mechanic rating.

(d) AIRCRAFT INSTRUMENT MAINTENANCE, MINOR REPAIRS,[11] AND MINOR ALTERATIONS. Maintenance, minor repairs, or minor alterations of an instrument must be made by one of the following:

A certificated repair station holding the appropriate rating.
The manufacturer of the instrument.

18.10–3 AGENCIES AUTHORIZED TO PERFORM MAJOR REPAIRS AND MAJOR ALTERATIONS. ***(CAA interpretations which apply to section 18.10.)***

(a) AIRCRAFT MAJOR REPAIRS AND MAJOR ALTERATIONS. Major repairs or major alterations of a certificated aircraft must be made by one of the following:

A certificated repair station holding the appropriate rating.
The manufacturer or the aircraft.
A certificated mechanic holding an aircraft mechanic rating.
A person under the direct supervision of a certificated mechanic holding an aircraft mechanic rating.

(b) AIRCRAFT ENGINE MAJOR REPAIRS AND MAJOR ALTERATIONS. Major repairs or major alterations of a certificated aircraft engine must be made by one of the following:

A certificated repair station holding the appropriate rating.

[10] See CAR 24.13 *Factory mechanic rating.*

[11] Replacement of removable instruments may be considered an aircraft minor repair (see CAM 18.6–1(g)) and may be made or supervised by a certificated mechanic.

The manufacturer of the aircraft engine.

A certificated mechanic holding an aircraft engine mechanic rating.

A person under the direct supervision of a certificated mechanic holding an aircraft engine mechanic rating.

(c) AIRCRAFT PROPELLER MAJOR REPAIRS AND MAJOR ALTERATIONS. Major repairs or major alterations of a certificated propeller must be made by one of the following:

A certificated repair station holding the appropriate rating.

The manufacturer of the propeller.

(d) AIRCRAFT INSTRUMENT MAJOR REPAIRS AND MAJOR ALTERATIONS. Major repairs or major alterations of an instrument must be made by one of the following:

A certificated repair station holding the appropriate rating.

The manufacturer of the instrument.

"CAR 18.11 ***Provisions for approval of major repairs and major alterations.*** **No aircraft, aircraft engine, or propeller which has undergone any major repair or major alteration shall be returned to service until examined, inspected, and approved by a duly authorized representative for the Administrator unless such repair or alteration has been executed in accordance with a manual or specification approved by the Administrator[2] and performed by a certificated repair station of the proper rating or by the manufacturer."**

"[2] Such manual or specification may, for example, be issued by the manufacturer, a certificated repair station, or by the Administrator. All such manuals or specifications issued by parties other than the Administrator must be approved by him."

"CAR 18.12 ***Flight tests.*** **When an aircraft or aircraft engine or propeller thereof has undergone a maintenance, minor repair, or minor alteration operation which may have changed its flight characteristics appreciably or substantially affected its operation in flight, or has undergone a major repair or major alteration, such aircraft shall, prior to carrying passengers, be test flown by a pilot having at least 200 solo hours and holding at least a private pilot certificate and appropriate rating for the aircraft to be test flown."**

RECORDING OF REPAIRS AND ALTERATIONS

"CAR 18.15 ***Minor repair and minor alteration logbook entries.*** **An adequate description of every minor repair or minor alteration of an aircraft, aircraft engine, or propeller shall be entered in the appropriate logbook over the signature and certificate number of the mechanic directly in charge of or performing such repair or alteration and in case a manufacturer or a certificated repair station makes said repair or alteration the appropriate logbook shall also be signed by an authorized official of such agency. The installation of an instrument in an aircraft shall be recorded in the aircraft logbook by the agency making the installation."**

18.15-1 SCOPE OF ENTRIES. ***(CAA policies which apply to section 18.15.)***

In case the work was performed in compliance with a request of the Civil Aeronautics Administration or in accordance with recommendation of the manufacturer of the unit repaired, or if replacement of a component purchased from the original manufacturer was involved, the entry should so state.

"CAR 18.16 ***Major repair and major alteration records.*** **A repair agency performing a major repair or major alteration on an aircraft, aircraft engine, or propeller, shall execute such Repair and Alteration Forms as may be prescribed and furnished by the Administrator, and shall deliver a duplicate copy of any such Form to the owner of the aircraft and make proper entries on the appropriate page of the Aircraft Operation Record."**

18.16-1 NUMBER OF FORMS REQUIRED. ***(CAA policies which apply to section 18.16.)***

The Repair and Alteration Form[12] should be submitted in duplicate unless the Aviation

[12] See Appendix A.

Safety agent requests additional copies. One copy of these forms should be kept by the owner of the aircraft, and the original should be given to the agent. In cases where an air carrier is operating under the provisions of an approved weight control system, the agent will determine that the proper entries are made in the pertinent weight record and that the effects of the weight changes have been accounted for. In all other cases the agent will ascertain that the proper loading schedule notations, overlays, or revised schedules have been prepared and submitted, if applicable.

18.16-2 SCOPE OF DATA ***(CAA policies which apply to section 18.16.)***

The Repair and Alteration Form should incorporate such technical data as are necessary to substantiate the airworthiness of the repair or alteration, either by reference to the figures and tables of this manual or by incorporating in the form pertinent stress analyses, weight and balance computations, test reports, drawings, well-dimensioned detail sketches or photographs. When these data are extensive, they should be appended to the original copy of such form, which is given to the Aviation Safety agent and referred to in both copies of the form.

18.16-3 AIR CARRIER FLEET REPAIRS AND ALTERATIONS. ***(CAA policies which apply to section 18.16.)***

In the case of simultaneous repairs or alterations being completed on several aircraft of an air carrier, only one complete file need be submitted. The Repair and Alteration Form should contain the identification marks and serial numbers of all aircraft affected. If additional aircraft of the same fleet are subsequently similarly repaired or altered, the Repair and Alteration Forms for these aircraft need merely refer to the previously submitted file by identification mark, serial number, and date. The agent will also determine that the operator has a reliable method of recording cumulative weight changes.

18.16-4 USE OF PHOTOGRAPHS. ***(CAA policies which apply to section 18.16.)***

Photographs, accompanied by detail descriptions of the work performed and the materials used and identified on these photographs, may be substituted for the usual sketches required on the Repair and Alteration Form. In this manner it is possible to record the original or damaged structure before any work on the structure is attempted and then record the various steps of the repair and alteration as the work progresses to final completion.

18.16-5 PROOF OF STRENGTH. ***(CAA policies which apply to section 18.16.)***

(a) COMPARISON WITH ORIGINAL PARTS. The original parts removed from an aircraft in making a major repair or alteration should be retained by the repair agency until the agent has completed his examination of the work for conformity or, in the case of a repair, satisfies himself that the repaired structure is airworthy.

(b) COMPARISON WITH MANUFACTURER'S DRAWING. The Aviation Safety agent may at his discretion require that he be furnished drawings from the original manufacturer of the part or parts in question. To obtain the necessary drawings, the repair agency should inform the manufacturer of the serial number of the airplane and the extent of the damage. Upon completion of the work, the drawings and executed Repair and Alteration Form, together with a copy of the manufacturer's letter of transmittal, should be presented to the agent to aid him in his inspection. In the letter of transmittal the manufacturer should list the drawings and specify the serial number of the airplane for which they were issued. When it can be shown that factory drawings are not obtainable, as in the case when the manufacturer is out of business and has no active successor, the repair agency may obtain photostats of such drawings from blueprinting firms at regular rates, through the Aircraft Division of the CAA regional office in the area.

18.16-6 PROOF OF MATERIAL CONFORMITY. ***(CAA policies which apply to section 18.16.)***

(a) APPROVED MATERIAL STATEMENT. A statement by the repair agency to the effect that the parts or materials used in the work have been purchased in accordance with approved specifications from the original manufacturer or a reputable aircraft supply firm will be required. Equally acceptable are invoices issued by the original manufacturer or his authorized agent.

(b) PROOF BY SAMPLE OR TEST. The Aviation Safety agent, at his discretion, may require that he be furnished with samples of the materials used in making the major repair or alteration. When it is not possible to compare the materials by a visual inspection or simple tests such as taking Rockwell or Brinell hardness readings, the results of material specification tests of the original and replacement samples should be supplied.

18.16–7 PROCEDURES AND GUIDING COMMENTS COVERING TYPICAL ALTERATIONS. ***(CAA policies which apply to section 18.16.)***

Detail procedures to be followed covering typical major alterations, such as the following cases, will be found in CAM 04.061.

The installation of an engine of a type other than that covered by a type (or approved type) certificate.

The installation of a tail wheel and tire in a previously approved tail skid installation.

The conversion of an approved type landplane or seaplane to approved skiplane status.

Guiding comments on the following typical alterations are given below.

(a) INCREASE IN GROSS WEIGHT. An increase in the gross weight will naturally require that the structure be able to withstand greater loads in flying and landing. There is usually involved, therefore, a study of the original design data and the preparation of a partial strength analysis, preferably by the manufacturer of the airplane. Likewise, the flying characteristics will be affected, so flight tests are usually required. Increases in weight are often accompanied by changes in weight distribution, the effects of which are covered in CAM 18.16–7 (b). Increases in gross weight for specific industrial purposes, such as crop dusting, may be permitted, but the aircraft will be restricted to that type of operation and will be identified thereon with the word "Restricted". An Aviation Safety agent should be contacted when an aircraft is to be used in restricted operations.

(b) CHANGE IN WEIGHT DISTRIBUTION. Any change in the location of items having considerable weight or the addition of new items (equipment, etc.) may have serious effects on the flight behavior of an airplane. When any changes in weight distribution are made, it is the repair agency's responsibility to determine, by computation or reweighing, whether or not the approved limits, which appear on the airplane specification in the case of later models, will be exceeded. If they are, approval cannot be granted unless compliance with all flight requirements is proved by means of a flight test.

It is of the greatest importance to realize that flight characteristics not only become worse gradually with rearward displacement of the center of gravity, but that a condition sometimes exists or will *finally* be reached *where a small change will have very large effects.* For this reason great care should be taken not to increase the weight of the airplane to the rear of the wing beyond that which was originally approved. Decreasing the weight carried forward of the wing, such as using a lighter propeller, will have a similar effect. Increasing the weight forward of the wing will tend to disturb the balancing in flight and might make the landing conditions dangerous. Naturally, the effects of weight changes will be greater nearer the nose and the tail. Reasonably small changes within the portion covered by the wing are not usually serious from a balancing or stability standpoint. For details regarding weight and balance procedure, see CAM 18.20–15.

(c) INSTALLATION OF NEW ITEMS. In addition to the effects on weight and weight distribution discussed in CAM 18.16–7 and 18.16–7 (a), there is a danger that a piece of new equipment, if improperly installed, will cause local loads which might seriously damage the airplane structure. It should be remembered that in flight maneuvers and in landing it is possible to develop inertia forces such that an item will impose a load of several times its own weight on the supporting structure. For instance, a 35-pound storage battery supported by a fuselage cross tube will have an effective weight of considerably over 100 pounds in hard landing. If the cross tube was not originally designed for this load, it will probably fail or bend. The greatest danger arises when such a partial failure occurs in landing, without being noticed, as the structure might then fail completely during some subsequent flight.

(d) POWERPLANT CHANGES. The powerplant installation on a certificated aircraft has been thoroughly checked in accordance with rules based on a study of accident and service records covering several years of operation of all types of aircraft. These records show that many accidents are due to improper installation in *small details* concerning the fuel or oil system, cowling, manifolding, and items of a like nature. It is, therefore, unwise to make any changes without careful consideration.

When changes appear to be necessary in the powerplant installation, the manufacturer of the aircraft should be consulted. If the manufacturer has no data concerning such a change, an Aviation Safety agent should be consulted to determine if the airworthiness of the airplane will be adversely affected by making the change.

With reference to propeller changes, the placard accompanying the certificate states the type of propeller which it is safe to use. A larger diameter propeller should not be used without investigation, because it may result in unsatisfactory ground clearance with resultant excessive wear on the tips. Furthermore, a change to a propeller of appreciably different diameter, either larger or smaller, or different pitch, might in some cases result in unsafe performance.

(e) CONTROL SURFACE CHANGES. Any change in the size of control surfaces affects the loading conditions for the airplane structure and therefore requires additional strength analyses, static tests, or both. Flight tests are usually required also. The owner is particularly warned against making minor changes on control surfaces, since the original design often just meets certain requirements for flutter prevention. No *balancing weights* should be removed or added without consulting the manufacturer and finally obtaining the agent's approval. In particular, it is essential that nothing be done to alter the contour of the nose section ahead of the hinge line or to increase the weight of movable surfaces to the rear of the hinge line. Balancing and trimming tabs have very powerful effects and should not be altered or allowed to become loosened. *All these precautions against flutter become increasingly important as the speed of the airplane is increased.* On high-speed airplanes any change of the control surfaces or system may result in flutter or dangerous vibration.

(f) FAIRING AND COWLING MODIFICATIONS. Although changes in fairing can usually be made without impairing the airworthiness of the aircraft, it has been found that certain airplanes are very sensitive to slight changes in body lines, windshield designs, and filleting. Obviously, any change in engine cowling affects the cooling and thereby introduces possibilities of malfunctioning or failure of the engine.

Low wing airplanes are usually supplied with wing-fuselage fillets which prevent tail buffeting. Any alteration of such fillets may be dangerous. The manufacturer should be consulted regarding such changes.

(g) APPLIANCE INSTALLATIONS. The aircraft and appliance manufacturer's instructions for installation of equipment should be closely followed. In particular, the recoil from flares should be provided for and, in the case of position lights, where certain angular limits are required, the light manufacturer's mounting instructions should be followed.

(h) BATTERIES. Batteries should be installed in accordance with the instructions contained in Civil Air Regulations 3.682 through 3.684, CAR 4.571, and CAR 4b.727 through 4b.730. It should be noted that dry batteries are not considered satisfactory for the operation of position lights.

18.16-8 DISPOSITION OF DATA. *(CAA policies which apply to section 18.16.)*

The repair agency should request an Aviation Safety agent to examine the work, the technical data, and the executed forms. In order to enable the agent to examine the work, it will generally be necessary to contact him while the work is in progress or arrange to have sufficient inspection openings in order that he may accomplish this inspection after completion of the work. Welds, particularly in the primary structure, should not be painted over until after the inspection unless the agent's permission is obtained. Careful consideration of the following check lists will eliminate many questions and the attendant delays:

(a) REPAIRS.

(1) Are the new parts made of the same material as the replaced parts? Is a stronger material used? Does the Repair and Alteration Form clearly state these facts?

(2) Are the new parts exact duplicates of the originals (as regards dimensions, fillets, welds, etc.)? Does the Repair and Alteration Form so state? If there are deviations, are they *all* listed and are all necessary new dimensions given?

(3) Does the Repair and Alteration Form state whether or not the original part was heat-treated? Does the Repair and Alteration Form state whether or not the new (or repaired) part is similarly heat-treated (or reheat-treated)?

(4) Is there any weight change? Does the Repair and Alteration Form state definitely one way or the other?

The above-mentioned items can usually be answered quickly by the man doing the work on the airplane, but their omission is very common.

(b) ALTERATIONS.

(1) How much does the article installed (or removed) weigh?

(2) What is the location of the removed or installed item? This should be given in inches from a datum.

(3) Has the balance of the airplane been substantiated for the alteration?

(4) How is the new item attached? Is the attachment shown in the sketches?

(5) Is the new item adequately described; i. e., are the manufacturer and model noted?

(6) If the new item is a structural alteration or involves attachment by means of additional structure, are all necessary dimensions and other engineering information given? Has the strength of the structure been adequately substantiated or shown to be obviously satisfactory?

18.16-9 AIRCRAFT OPERATION RECORD ENTRIES. *(CAA policies which apply to section 18.16.)*

A repair agency engaged in the major repair or major alteration of an aircraft, aircraft engine, or propeller should also make an entry descriptive of the work done on the appropriate page of the Aircraft Operation Record. It is urged that in all questionable repair or alteration cases, which might involve new repair methods, flight characteristics, or complicated questions of weight and balance, the executed forms, together with all data, be submitted to the regional offices of the Aircraft Division of the Civil Aeronautics Administration for approval of the questionable points prior to final completion of the work.

18.16-10 SPECIAL RECORDING FOR CERTIFICATED REPAIR STATIONS AND MANUFACTURERS. *(CAA policies which apply to section 18.16.)*

In order to maintain the records of all work done on aircraft under the special privilege provisions contained in CAR 18.11, certificated repair stations (other than those working on certificated air-carrier aircraft) and manufacturers should:

(a) Supply the owner of the repaired or altered aircraft with a copy of the Civil Aeronautics Administration's Repair and Alteration Form and append to it all details such as drawings, balance investigations, or weight records, as the case may be.

(b) Make the necessary revisions to the

pertinent page of the Aircraft Operations Record.

"CAR 18.17 ***Provision for air carrier records.*** **Logbook and aircraft operation record entries required in this part may be replaced, in the case of repairs or alterations to scheduled air carrier aircraft, by a suitable system of recording repairs, alterations, and signatures of responsible personnel."**

18.17-1 REPAIR BASE RECORDS. ***(CAA policies which apply to section 18.17.)*** The special procedures outlined in CAM 18.16-9 need not be followed by repair stations or certificated air carriers repairing or altering air carrier aircraft. The repair base records, including the weight-control records maintained by these agencies, are interpreted to serve the identical purpose.

DESIGN, TECHNIQUES, AND MATERIALS

"CAR 18.20 ***Design, techniques, and materials.*** **Repairs shall be so executed, and materials of such strength and quality shall be used that the condition of the repaired aircraft, aircraft engine, propeller, or instrument shall be at least equivalent to its original or a properly altered condition in regard to aerodynamic and mechanical function, structural strength, and resistance to vibration and deterioration, and all other qualities affecting airworthiness. Alterations shall be so designed and executed that the altered aircraft, aircraft engine, propeller, or instrument will comply with the airworthiness requirements in effect when the particular model of the aircraft or part of the aircraft was originally certificated and, in addition, with particular provisions of the current airworthiness requirements rendered necessary for safe operation by the alteration."**

18.20-1 WOOD AIRCRAFT STRUCTURES.[13] ***(CAA policies which apply to section 18.20.)***

(a) MATERIALS USED IN WOOD CONSTRUCTION. Three general forms of wood are used in aircraft: solid wood, plywood, and laminated wood. In addition, several kinds of modified wood are being used for special purposes. For general repair work, however, the three forms listed above are of principal importance as they constitute the bulk of all wood aircraft construction materials.

(1) QUALITY OF WOOD. All wood and plywood used in the repair of aircraft structures should be of aircraft quality. Table 1-1 lists the permissible variations in characteristics and properties of aircraft wood.

(2) SPECIES SUBSTITUTION. The species used to repair a part should be the same as that of the original whenever possible; however, permissible substitutes are given in Table 1-1.

(3) EFFECTS OF SHRINKAGE. When the moisture content of a piece of wood is lowered, its dimensions decrease. The dimensional change is greatest in a tangential direction (across the fibers and parallel to the growth rings), somewhat less in a radial direction (across the fibers and perpendicular to the growth rings), and is negligible in a longitudinal direction (parallel to the fibers).

These dimensional changes can have several detrimental effects upon a wood structure such as loosening of fittings and wire bracing, and checking or splitting of wood members.

A few suggestions for minimizing these shrinkage effects are:

(i) Use of bushings that are slightly short so that when the wood member shrinks, the bushings do not protrude and the fittings may be tightened firmly against the member.

(ii) Gradual dropping off of plywood face plates either by feathering or by shaping as shown in Figure 1-1.

(4) REPLACEMENT OF DRAIN HOLES AND SKIN STIFFENERS. Whenever repairs are made that require replacing a portion that included drain holes, skin stiffeners, or any other items, the repaired portion must be provided with similar drain

[13] Note: Complete information on the general design and fabrication of wood aircraft structures may be found in ANC-18, "Design of Wood Aircraft Structures," and ANC-19, "Wood Aircraft Inspection and Fabrication," copies of which may be obtained for 75c and $1.00, respectively, from the Superintendent of Documents, Government Printing Office, Washington 25, D. C.

holes, skin stiffeners, or items of the same dimensions in the same location. Reinforcing, under skin repairs, that interferes with the flow of water from some source, such as inspection holes, must be provided with drain holes at the lowest points.

(5) FLUTTER PRECAUTIONS. When repairing control surfaces, especially on high-performance airplanes, care should be exercised that the repairs do not involve the addition of weight aft of the hinge line. Such procedure may adversely disturb the dynamic and static balance of the surface to a degree which would induce flutter. As a general rule it will be required to repair control surfaces in such a manner that the structure is identical to the original so that the weight distribution is not affected in any way.

(6) GLUES AND GLUING. Satisfactory glue joints in aircraft will develop the full strength of wood under all conditions of stress. To produce this result the gluing operation must be carefully controlled so as to obtain a continuous, thin, uniform film of solid glue in the joint with adequate adhesion to both surfaces of the wood. Some of the more important conditions involve:

Properly prepared wood surfaces.
Glue of good quality, properly prepared.
Good gluing technique.

(i) PREPARATION OF WOOD SURFACES FOR GLUING. It is recommended that no more than eight hours be permitted to elapse between final surfacing and gluing. The gluing surfaces should be machined smooth and true with planers, jointers or special miter saws. Planer marks, chipped or loosened grain, and other surface irregularities should not be permitted. Sandpaper should normally never be used to smooth wood glue surfaces. Satisfactory sawed surfaces should approach well-planed surfaces in uniformity, smoothness, and freedom from crushed fibers.

Tooth-planing, or other means of roughening smooth, well-planed surfaces of normal wood before gluing are not recommended.

Such treatment of well-planed wood surfaces may result in local irregularities and objectionable rounding of edges. While sanding of planed surfaces is not recommended for soft woods, sanding is a valuable aid in improving the gluing characteristics of some hard plywood surfaces; wood that has been compressed through exposure to high pressures and temperatures; resin-impregnated wood (impreg and compreg); and laminated paper plastic (papreg).

Wood surfaces for gluing should be free from oil, wax, varnish, shellac, lacquer, enamel, dope, sealers, paint, dust, dirt, old glue, crayon marks and other extraneous materials.

Wetting tests are useful as a means of detecting the presence of wax. Drops of water placed on the surface of wax-coated plywood do not spread or wet the wood. At present, preliminary gluing tests appear to be the only positive means of actually determining the gluing characteristics of plywood surfaces.

(ii) GLUES. Glues used in aircraft repair fall into two general groups: Casein glues, and resin glues. Any glue that meets the performance requirements of applicable Army or Navy specifications is satisfactory for use in certificated civil aircraft. In all cases glues are to be used strictly in accordance with the glue manufacturers' recommendations.

(*a*) CASEIN GLUES. Casein glues are probably more widely used than any of the resin glues in aircraft repair work. The forms, characteristics, and properties of water-resistant casein glues have remained substantially the same for many years except for the addition of preservatives. Casein glues for use in aircraft should contain suitable preservatives such as the chlorinated phenols and their sodium salts, to improve their resistance to organic deterioration under high humidity exposures. Most casein glues are sold in powder form ready to be mixed with water at ordinary room temperatures.

(*b*) SYNTHETIC RESIN GLUES. Synthetic resin glues for wood are outstanding in that they retain their strength and durability under moist conditions and even after exposure to water. The best-known and most commonly used synthetic resin glues are the phenol-formaldehyde, resorcinol-formaldehyde, and urea-formaldehyde types. Materials, such as walnut-shell flour or wood flour, are often added by the glue manufacturer to the resin glues to give better working characteristics and joint-forming properties. Table 1-2 has been prepared as an aid in the selection of cold-setting synthetic resin glues. It has been derived largely from the glue manufacturers' directions and instructions, the experience of the users of glues, and such test results as are available. This list is incomplete and subject to change as some brands of glues are discontinued, others modified, and new glues developed and marketed. The inclusion of any glue in this list does not constitute an endorsement on the part of any Government agency or assurance that it will meet applicable specifications.

The glues listed in this table are the room-temperature setting type. The suitable curing temperatures for the urea-formaldehyde type vary from 70° F. to 75° F., and for the resorcinol glues from 70° F. up. The strength of the joint cannot be depended upon if assembled and cured at temperatures below 70° F. Among the more common cold-press synthetic glues which require a curing temperature of at least 70° F. are Casco Resin Liquid No. 5, Cascamite ANS, Bakelite BCV-12772, Lauxite 77X, Perkins DC-246, Plaskon 250-2, Uformite CB-551, Weldwood, etc. Other types available for use, such as the intermediate temperature and hot-press types, are not presented in the table as their use generally requires special equipment and special quality control.

(*c*) MIXING OF RESIN GLUES. Liquid resin glues may come ready for use or in a form which requires only the addition of a hardener. In all cases the mixing, glue consistency, assembly time, etc., should comply with the glue manufacturer's recommendations and instructions. Cold-setting, synthetic-resin glues, when prepared for use, are usually sharply limited in working life, and care should be taken to discard the glue and clean the equipment before the end of the working-life period. In very warm weather it may be found advisable to keep the glue pot in a bath of cool water, approximately 70° F., to prolong the working life of the mixture.

(iii) GLUING TECHNIQUE.

(*a*) SPREADING OF GLUE. To make a satisfactory glue joint, glue should be spread evenly on both of the surfaces to be joined. It is recommended that a clean brush be used and care taken to see that all surfaces are covered. The spreading of glue on but one of the two surfaces is not recommended.

(*b*) ASSEMBLY TIME IN GLUING. Where pieces of wood are coated and exposed freely to the air, a much more rapid change in consistency of the glue occurs than where the pieces are laid together as soon as the spreading has been done. The condition of free exposure is conveniently referred to as "open assembly" and the other as "closed assembly."

When cold-setting glues are coated on wood parts and left exposed to the atmosphere (open assembly), the allowable assembly time is appreciably reduced compared with closed assembly periods. Approximate ranges in assembly times and gluing pressures should be as recommended by the glue manufacturer.

The pressing time for casein and resin glue joints should, in general, be seven hours or more. Other types of glue require various times and temperatures for curing. Glue joints increase in strength mainly as a result of drying; hence, where it is convenient to do so, it is better to maintain pressure from one day to the next. The longer pressing periods are desirable, as this enables the joints to reach a higher proportion of their final strength before being disturbed.

(*c*) GLUING PRESSURE. Pressure is used to squeeze the glue out into a thin continuous film between the wood layers, to force air from the joint, to bring the wood surfaces into intimate contact with the glue, and to hold them in this position during the setting of the glue.

Pressure should be applied to the joint before the glue becomes too thick to flow and is accomplished by means of clamps, presses, or other mechanical devices.

Nonuniform gluing pressure commonly results in weak and strong areas in the same joint. The amount of pressure required to produce strong joints in aircraft assembly operations may vary from 125 to 150 pounds per square inch for softwoods and 150 to 200 pounds per square inch for hardwoods. Insufficient pressure and poorly machined wood surfaces usually result in thick glue lines which form a weak joint and should be carefully guarded against.

(*d*) METHOD OF APPLYING PRESSURE. The methods employed in applying pressure to joints in aircraft gluing operations range from the use of brads, nails, screws, and clamps to the use of hydraulic and electric power presses. Hand nailing is used rather extensively in the gluing of ribs and in the application of plywood skins to the wing, control surfaces, and fuselage frames.

On small joints such as found in wood ribs, the pressure is usually applied only by nailing the joint gussets in place after spreading the glue. Since small nails must be used to avoid splitting, the gussets must be comparatively large in area to compensate for the relative lack of pressure. Nail spacing should be at least four per square inch and should in no event be less than one nail per ¾ inch. Small brass screws may also be used advantageously where the particular parts to be glued are relatively small and do not allow application of pressure by means of clamps.

Spar splices should always be clamped by means of cabinet-maker's parallel clamps or similar types. Handspring clamps should be used in conjunction with softwood only. Due to their limited pressure area, they must be applied with a pressure-distributing strip or block at least twice as thick as the member to be pressed.

(b) SCARF JOINTS

(1) GENERAL. The scarf joint is the most satisfactory method of making a joint in the grain direction between two solid wood members. Both parts should be cut accurately because the strength of the joint depends upon maximum contact between the surfaces being glued.

(2) GRAIN DIRECTION. The scarf cut should be made in the general direction of the grain slope as shown in Figure 1-2. (See figure for note concerning allowable deviation from grain direction.)

(c) SPARS.

(1) SPLICING OF SPARS. A spar may be spliced at any point except under wing attachment fittings, landing gear fittings, engine-mount fittings, or lift and interplane strut fittings. These fittings should not overlap any part of the splice. Splicing under minor fittings such as drag wire, antidrag wire or compression strut fittings is acceptable under the following conditions:

The reinforcing plates of the splice should not interfere with the proper attachment or alignment of the fittings. The locations of pulley support brackets, bellcrank support brackets or control surface support brackets should not be altered.

The reinforcing plate may overlap drag or antidrag wire or compression strut fittings if the reinforcing plates are on the front face of the front spar or on the rear face of the rear spar. In such cases it will be necessary to install slightly longer bolts. The inside reinforcing plate should not overlap drag strut fittings unless such overlapping does not require sufficient shortening of compression struts or changes in drag truss geometry to prevent adjustment for proper rigging. Even though take-up is sufficient, it may be necessary to change the angles on the fittings. Splices should be spaced so that they do not overlap. Acceptable methods of splicing the various types of spars are shown in Figures 1-3 through 1-6. Reinforcing plates must be used as indicated on all scarf repairs to spars and the slopes of scarfs shown are minimum slopes.

(2) SPLICING OF BOX SPAR WEBS. The method of splicing plywood webs is shown in Figure 1-7. Plywood webs should always be spliced and reinforced with the same type of plywood. Solid wood should never be used to replace plywood webs as plywood is stronger in shear than solid wood of the same thickness. The face grain of plywood replacement webs and reinforcing plates should be in the same direction as the original member to insure that the new web will have the required strength.

(3) REPLACING SOLID-TYPE SPARS WITH LAMINATED TYPE. Solid spruce spars may be replaced with laminated ones or vice versa, provided the material is of the same high quality. External plywood reinforcements should always be replaced with plywood as in the original structure.

(4) LONGITUDINAL CRACKS AND LOCAL DAMAGE. Cracked spars (except box spars) may be repaired by gluing plates of spruce or plywood of sufficient thickness to develop the longitudinal shear strength of the spar to both sides of it. Such plates should extend well beyond the termination of the cracks as shown in Figure 1-8. A method of repairing small local damage to either the top or bottom side of a spar is also shown in this figure.

(i) LONGITUDINAL CRACKING OF WOOD WING SPARS IN AIRPLANES OPERATING IN ARID REGIONS. Airplanes having wood spars and being operated in arid regions may develop longitudinal spar cracks in the vicinity of the plywood reinforcing plates. These cracks result from the tendency of the spar to shrink when drying takes place. Plywood resists this tendency and causes a cross-grain tensile failure in the basic spar. Cracks start under the plywood plates; usually, but not necessarily, at a bolt hole or cut-out and spread in each direction until, in most cases, they extend a short distance beyond the ends of the plates where the resistance to spar shrinkage disappears. Other factors which have been found conducive to the formation of cracks due to spar shrinkage in the region of plywood plates are poor protective finishes, large cut-outs, and metal fittings which utilize two lines of large diameter bolts.

The presence of cracks does not necessarily mean that the spar must be discarded. If the crack is not too long or too close to either edge and can be reinforced properly, it will probably be more economical and satisfactory to effect a repair than to install a new spar or section. However, a generally acceptable procedure suitable for all airplane models cannot be described here. In view of the possible structural problems involved, it is recommended that the manufacturer or the Civil Aeronautics Administration be contacted for specific instructions before making repairs not in accordance with the manufacturers' approved instructions or the recommendations of this manual, because of the possibility of strength deficiencies.

(5) ELONGATED HOLES. In cases of elongated bolt holes in a spar or cracks in the vicinity of bolt holes, a new section of spar should be spliced in or the spar replaced entirely, unless the method of repair is specifically approved by a representative of the Civil Aeronautics Administration. In many cases it has been found advantageous to laminate the new section of the spar (using aircraft plywood for the outer faces), particularly if the spar roots are being replaced.

(d) RIBS.

(1) GENERAL. Complete ribs should be made from a manufacturer's approved drawing or from a drawing made by the repair agency and certified by the manufacturer as correct, except that the original rib may be used as a pattern in making the new rib if it is not too seriously damaged to permit comparison. Wood ribs should preferably not be attached to wood spars by nails driven through the rib cap strips, as this weakens the rib materially. The attachment should be by means of glue, with cement-coated, barbed or spiraled nails driven through the vertical rib members on each side of the spar. The drawing or pattern should be retained by the repair agency for

use by the Aviation Safety agent in making his inspection.

(2) REPAIRS OF WOOD STRUCTURES AT A JOINT, BETWEEN JOINTS, AT TRAILING EDGES, OR AT SPARS. Acceptable methods of repairing damaged ribs are shown in Figure 1-9.

(3) COMPRESSION RIBS. Acceptable methods of repairing damaged compression ribs are shown in Figure 1-10. (A) illustrates the repair of a compression rib of the "I" section type, i. e., wide, shallow cap strips, a center plywood web, with a rectangular compression member on each side of the web. The rib is assumed to be cracked through cap strips, web member, and compression member. Cut the compression member as shown in (D), remove, and replace the shortest section, adding the reinforcing blocks as also shown in (D). Cut and replace the aft portion of the cap strips, reinforcing as shown in Figure 1-9, except that the reinforcing blocks are split in the vertical direction to straddle the center web. The plywood side plates, as indicated in (A), are glued on. These plates are added to reinforce the damaged web. (B) illustrates a compression rib of the type that is basically a standard rib with rectangular compression members added to one side and a plywood web to the other side. The method used in this repair is essentially the same as in (A) except that the plywood reinforcing plate shown solid black in section B-B is continued the full distance between spars. (C) illustrates a compression rib of the "I" type with a rectangular vertical member each side of the web. The method of repair is essentially the same as in (A) except that the plywood reinforcing plates on each side shown in solid black in section C-C are continued, as in (C), the full distance between spars.

(e) PLYWOOD SKIN.

(1) GENERAL. Extensive repairs to damaged stressed skin plywood structures should be made in accordance with specific recommendations from the manufacturer. It is recommended that repairs be made by replacing the entire panel from one structural member to the next if damage is very extensive. When damaged plywood skin is repaired, the adjacent internal structure should be carefully inspected for possible hidden damage. Any defective frame members should be repaired prior to making skin repairs.

(i) TYPES OF PATCHES. Four types of patches—the surface or overlay patch, the splayed patch, the plug patch, and the scarf patch—are acceptable for repairing plywood skins. Surface patches should not be used on skins over ⅛ inch thick. Splayed patches should not be used on skins over 1/10 inch thick. There are no skin thickness limitations for the use of scarf patches and plug patches.

(ii) DETERMINATION OF SINGLE OR DOUBLE CURVATURE. Much of the outside surface of plywood aircraft is curved. On such areas, plywood used for repairs to the skin must be similarly curved. Curved skins are either of single curvature or of double (compound) curvature. A simple test to determine which type of curvature exists may be made by laying a sheet of heavy paper on the surface in question. If the sheet can be made to fit the surface without wrinkling, the surface is either flat or has single curvature. If, however, the sheet cannot be made to fit the surface without wrinkling, the surface is of double curvature.

(iii) REPAIRS TO SINGLE CURVATURE SKIN. Repairs to skins of single curvature may usually be formed from flat plywood, either by bending it dry or after soaking it in hot water. The degree of curvature to which a piece of plywood can be bent will depend upon the direction of the grain and the thickness. Table 1-3 is presented as a guide in determining which process of bending should be used for the curvature being considered.

Plywood after softening may be bent on a cold ventilated form or it may be bent over the leading edge near the part being patched if space permits. In either method it should be allowed to dry completely on the form. When bending plywood over a leading edge, drying may be hastened by laying a piece of coarse burlap over the leading edge before using it as a bending form. A fan to circulate the air over the bent piece will speed the drying. In bending pieces of small radii or to speed up the bending of a large number of parts of the same curvature, it may be necessary to use a heated bending form. The surface temperature of this form may be as high as 149° C. (300° F.), if necessary, without danger of damage to the plywood. The plywood should be left on the form, however, only long enough to dry to room conditions.

(iv) REPAIRS TO DOUBLE CURVATURE SKIN. The molded plywood necessary for a repair to a damaged plywood skin of double curvature cannot be made from flat plywood unless the area to be repaired is very small or is of exceedingly slight double curvature; therefore molded plywood of the proper curvature must be on hand before the repair can be made. If molded plywood of the proper curvature is available, the repair may be made following the recommended procedures.

(2) SPLAYED PATCH. Small holes with largest dimensions not over 15 times the skin thickness, in skins not more than 1/10 inch in thickness, may be repaired by using a circular splayed patch as illustrated in Figure 1-11. The term splayed is used to denote that the edges of the patch are tapered but the slope is steeper than is allowed in scarfing operations. The following steps should be taken in making a splayed patch:

(i) Lay out the patch according to Figure 1-11. Center the dividers as near to the damage as is possible or tack a small piece of plywood over the hole for a center point and draw two circles, the inner one to be the size of the hole and the outer one marking the limits of the taper. The difference between the radii is 5T (5 times the thickness of the skin). If one leg of the dividers has been sharpened to a chisel edge, the dividers may be used to cut the inner circle completely through.

(ii) Taper the hole evenly to the outer circle with a chisel, knife, or rasp.

(iii) Prepare a circular tapered patch to fit the prepared hole, and glue the patch into place with face grain direction matching that of the original surface.

(iv) Use waxed paper between the patch and a plywood pressure plate cut to the exact size of the patch. This prevents extruded glue from binding patch and plate together. Center the plate carefully over the patch.

(v) Apply pressure. As there is no reinforcing behind this patch, care must be used so that pressure is not great enough to crack the skin. On horizontal surfaces, weights or sandbags will be sufficient. On vertical surfaces apply hand clamps lightly but snugly. On patches too far in for the use of standard hand clamps, jaws of greater length may be improvised.

(vi) Fill, sand, and refinish the patch.

(3) SURFACE PATCH. Plywood skins that are damaged between or along framing members may be repaired by surface or overlay patches as shown in Figure 1-13. The damaged skin should be trimmed to a rectangular or triangular shape and the corners rounded. The radius of rounded corners should be at least five times the skin thickness. Surface patches should best be covered with fabric before finishing. Fabric should overlap the original skin at least 2 inches. Surface patches located entirely aft of the 10 percent chord line or which wrap around the leading edge and terminate aft of the 10 percent chord line are permissible. Patches located entirely aft of the 10 percent chord line should have their forward edges beveled to four times the skin thickness. Surface patches may cover as much as one frame (or rib) space and a 50-inch perimeter. The face grain direction should be the same as the original skin.

(4) SCARF PATCH.

(i) GENERAL. A properly prepared and inserted scarf patch is the best repair for damaged plywood skins. It is the preferred type for most skin repairs. Figure 1-14 shows the details and dimensions to be used when installing typical scarf skin patches when the back of the skin is accessible.

Figure 1-15 should be followed when the back of the skin is not accessible. The scarf slope of 1 in 12 shown in both figures is the steepest slope permitted for all species of plywood. If the radius of curvature of the skin at all points on the trimmed opening is greater than 100 times the skin thickness, a scarf patch may be installed.

Scarf cuts in plywood may be made by hand plane, spoke shave, scraper or accurate sandpaper block. Rasped surfaces, except at the corners of scarf patches, and sawed surfaces are not recommended as they are likely to be rough or inaccurate.

Nail strip gluing is often the only method available for gluing scarf joints in plywood when used in repair work; therefore it is essential that all scarf joints in plywood be backed with plywood or solid wood to provide adequate nail-holding capacity. The face grain direction of the plywood patch should be the same as that of the original skin.

(ii) SCARF PATCHES (BACK OF SKIN ACCESSIBLE). When the back of a damaged plywood skin is accessible (such as a fuselage skin), it should be repaired with scarf patches following the details shown in Figure 1-14. Whenever possible the edge of the patch should be supported as shown in section C-C. When the damage follows or extends to a framing member, however, the scarf may be supported as shown in section B-B.

Damages that do not exceed 25 times the skin thickness in diameter after being trimmed to a circular shape, and if the trimmed opening is not nearer than 15 times the skin thickness to a framing member, may be repaired as shown in Figure 1-14, section D-D. The backing block is especially shaped from solid wood and fitted to the inside surface of the skin, and is temporarily held in place with nails. A hole, the exact size of the inside circle of the scarf patch, is made in the block and is centered over the trimmed area of damage. The block is removed after the glue on the patch has set, and leaves a flush surface to the repaired skin.

(iii) STEPS IN MAKING SCARF PATCH (BACK OF SKIN NOT ACCESSIBLE).

(*a*) After removing damaged sections, install backing strips, as shown in Figure 1-15, along all edges that are not fully backed by a rib or a spar. To prevent warping of the skin, backing strips should be made of a soft-textured plywood, such as yellow poplar or spruce, rather than solid wood. All junctions between backing strips and ribs or spars should have the end of the backing strip supported by a saddle gusset of plywood.

(*b*) If needed, nail and glue new gusset plate to rib. It may be necessary to remove and replace the old gusset plate by a new saddle gusset or it may be necessary to nail a saddle gusset over the original.

(*c*) Attach nailing strips to hold backing strips in place while the glue sets. Use bucking bar where necessary to provide support for nailing. Unlike the smaller patches made in a continuous process, work at the airplane must wait while the glue holding the backing strips sets. After setting, complete finishing in usual manner.

(5) PLUG PATCHES.

(i) GENERAL. Two types of plug patches, oval and round, may be used on plywood skins provided the damage can be covered by the patches whose dimensions are given in Figures 1-12 and 1-16. As the plug patch is strictly a skin repair it should be used only for damage that does not involve the supporting structure under the skin. Oval patches must be prepared with the face grain carefully oriented to the same direction as the original skin. Orientation of the face grain direction of the round plug patch to that of the skin surface is no problem, as the round patch may be rotated until grain directions match.

(ii) STEPS IN MAKING OVAL PLUG PATCH.

(*a*) Explore the area about the hole to be sure it lies at least the width of the oval doubler from a rib or a spar. Refer to Figure 1-12 for repair details.

(*b*) Lay a previously prepared oval plug patch over the damage and trace the patch. Saw to the line and trim the hole edges with a knife and sandpaper.

(*c*) Mark the exact size of the patch on one surface of the oval doubler and apply glue to the area outside the line. The oval doubler should be made of some soft-textured plywood, such as yellow popular or spruce. Insert doubler through the hole and bring it, glue side up, to the underside of the skin with its pencil outline of the patch matching the edges of the hole. If the curvature of the surface to be repaired is greater than a rise of ⅛ inch in 6 inches, the doubler should be preformed, by hot water or steam bending, to the approximate curvature.

(*d*) Apply nailing strips, outlining the hole, to apply glue pressure between doubler and skin. Use bucking bar to provide support for nailing. When two rows of nails are used, stagger nail spacing.

(*e*) Apply glue to remaining surface and to an equivalent surface on the patch.

(*f*) Lay the patch in position over the doubler and screw the pressure plate to the patch assembly using a small nail to line up the holes that have been previously made with patch and plate matching. No. 4 roundhead screws are used. Lead holes in the plywood doubler are not necessary. Waxed paper or cellophane between the plate and patch prevents glue from sealing the plate to the patch. No clamps or further pressure need be applied as the nailing strips and screws exert ample pressure. Hot sandbags, however, may be laid over the patch to speed the setting of the glue. Finish in the usual manner.

(iii) ROUND PLUG PATCH. The steps in making a round plug patch shown in Figure 1-16 are identical with those for making the oval patch except the insertion of the doubler. In using the round patch, where access is from only one side, the round doubler cannot be inserted unless it has been split.

(6) FABRIC PATCH. Small holes not exceeding 1 inch in diameter, after being trimmed to a smooth outline, may be repaired by doping a fabric patch on the outside of the plywood skin. The edges of the trimmed hole should first be sealed, and the fabric patch should overlap the plywood skin by at least 1 inch. Holes nearer than 1 inch to any frame member or in the leading edge, or frontal area of the fuselage should not be repaired with fabric patches.

(f) FINISHING STRUCTURAL REPAIRS.

(1) GENERAL. Any repair to spars, ribs, skin surfaces, or other structural parts of the airframes involves finishing as the final step in the job.

(2) PRECAUTIONS TO BE OBSERVED.

(i) When making repairs, avoid excessive contamination of surfaces with glue squeeze-out at joints and on all surfaces. Excess glue should always be removed before applying finish. Because paints and glues are incompatible, even a slight amount of glue underneath the finish may cause premature deterioration.

(ii) Soiling substances, such as oil and grease, should be removed as completely as possible. Naptha may be used to sponge off oil and grease. Markings that are made by grease pencils or lumber crayons containing wax are harmful and should be removed, but marks made by ordinary soft graphite pencils and non-blotting stamp-pad inks may be safely finished over. All dust, sander dust, dirt, and other solid particles should be cleaned off.

(iii) Sawdust, shavings, and chips should be removed from enclosed spaces before they are sealed off by replacement of skin. A vacuum cleaner is useful for such cleaning.

(iv) Since no satisfactory gluable sealer has yet been developed, it is necessary to avoid applying sealer over the areas where glue will be applied. Areas to receive glue should be marked off with pencil, allowing an additional ¼ inch on each side of the glue area to provide for misalignment when mating the parts. It is preferable to leave some unsealed areas rather than risk weakening the glue joint by accidental overlap of the sealer into the glued areas.

(v) Finish is likely to crack when applied over flush-driven nails and screws. To avoid this a strip of tape may be applied over the heads after application of sealer and before the final finish is applied.

(vi) Fill all holes left from nail-strip gluing or countersunk nails and screws with a wood filler before finishing the surface. It may be necessary to cover with a patching putty the slight depressions left after applying filler if a completely smooth surface is desired, but as a rule patching putty may safely be dispensed with.

(vii) Surfaces which are likely to come in contact with fabric during the doping process should be treated with a dope-proof paint, cellophane tape, etc., to protect them against the action of the solvents in the dope.

(3) FINISHING OF INTERIOR SURFACES. Repaired ribs, spars, interior of plywood skin, and other internal members, including areas of contact between metal and wood, should be finished by applying at least two coats of spar vanish. Built-up box spars and similar closed structures should be protected on the interior by at least one heavy coat of spar varnish or lionoil. Where better protection is required, as on the surfaces of wheel wells and the bottoms of hulls below the floor boards, an additional coat of aluminized sealer consisting of 12 to 16 ounces of aluminum paste per gallon of sealer may be applied.

(4) FINISHING OF EXTERIOR SURFACES. Exterior surfaces should first be sealed with at least two coats of sealer or spar varnish. The surface finish should then be completed by the application of enamel, aluminized varnish or other special finish as required to duplicate the original finish. If dope or lacquer is used to complete the finish, the sealer coats should be dope-proof. Spar varnish or sealer conforming to Specification AN–TT–V–116 is satisfactory.

(5) FINISHING OF END-GRAIN SURFACES. End-grain surfaces, such as edges of plywood skins and holes in spars and other primary structural members, require careful protection. Sand these surfaces smooth. Apply two coats of a highly pigmented sealer, or one coat of wood filler, and one coat of clear sealer to end-grain interior surfaces and cut holes. Exterior end-grain surfaces (except those covered with doped fabric) require an additional (third) coat of clear sealer. A final coat of aluminized varnish may be applied to end-grain surfaces. If the surfaces are to be finished with dope or lacquer, a dope-proof sealer similar to Specification AN–TT–V–116 should be used.

Exposed end-grain includes such surfaces as those around vent holes, inspection holes and fittings, and exposed scarfed or tapered surfaces such as those of tapered blocking.

(6) FINISHING WITH FABRIC OR TAPE.

(i) To refinish with fabric or tape, it is first necessary to insure that paint has been removed from an area greater than that to be covered by the fabric.

(ii) Apply two brush coats of a dope-proof sealer similar to Specification AN–TT–V–116, allowing the first coat to dry 2 hours and the second coat at least 6 hours. Follow with one coat of clear dope, and allow it to dry 45 minutes. Apply a second coat of clear dope and lay into the wet film a piece of pinked-edge glider or airplane cloth. All air bubbles should be worked out by brushing to insure maximum adherence. Allow this to dry 45 minutes. Apply one brush coat to insure proper penetration and at least one spray coat of clear dope, allowing each to dry 45 minutes. The dried spray coat may be scuffed with fine sandpaper to obtain a smoother finish. Complete the refinishing of the surface by application of lacquer, enamel, or aluminized varnish as required to match the adjacent areas.

(iii) The size of the fabric patch should be such as to extend at least ½ inch on each side of any crack or group of cracks, at least 1 inch on each side of a scarfed joint glue line, and at least 2 inches beyond any edge of a skin patch, to insure proper adhesion.

18.20–2 FABRIC COVERING. ***(CAA policies which apply to section 18.20.)***

(a) TEXTILE MATERIALS. All fabric, surface tape, reinforcing tape, machine thread, lacing cord, etc., used for re-covering or repairing an aircraft structure should be of high-grade aircraft textile material of at least as good quality and equivalent strength as those listed in subparagraphs (1) through (7).

(1) AIRCRAFT FABRIC. Acceptable fabrics for covering wings, control surfaces, and fuselages are listed in Table 2–1. Fabric conforming to the Automotive Material Specifications incorporate a continuous marking showing the specification number to permit identification of the fabric in the field.

(2) RE-COVERING AIRCRAFT WITH ORIGINAL TYPE FABRIC. Aircraft should be re-covered or repaired with fabric of at least as good quality and equivalent strength as that originally used on the aircraft.

(3) REINFORCING TAPE. Acceptable reinforcing tape is listed in Table 2–2. Reinforcing tape should be of similar quality to the fabric and at least one-half the strength of that conforming to Specification AN–DDD–T–91a, Type I.

(4) SURFACE TAPE. Surface tape (also finishing tape) should have approximately the same properties as the fabric used. See Table 2–2.

(5) LACING CORD. Lacing cord should have a strength of at least 80 pounds double or 40 pounds single strand. Acceptable lacing cord is listed in Table 2–2.

(6) MACHINE THREAD. Machine thread should have a strength of at least 5 pounds single strand (Table 2–2).

(7) HAND-SEWING THREAD. Hand-sewing thread should have a strength of at least 14 pounds single strand (Table 2–2).

(b) COVERING PRACTICES.

(1) GENERAL. The method of fabric attachment should be identical, as far as strength and reliability is concerned, to the method used by the manufacturer of the airplane to be re-covered or repaired. Fabric may be applied so that either the warp or fill threads are parallel to the line of flight. Either the envelope method or blanket method of covering is acceptable. (See CAM 18.20–2(b)(5).)

(2) FLUTTER PRECAUTIONS. When repairing control surfaces, especially on high-performance airplanes, care should be exercised that the repairs do not involve the addition of weight aft of the hinge line. Such procedure may adversely disturb the dynamic and static balance of the surface to a degree which would induce flutter. As a general rule, it will be required to repair control surfaces in such a manner that the structure is identical to the original so that the weight distribution is not affected in any way.

(3) PREPARATION OF THE STRUCTURE FOR COVERING. In covering work, one of the most important items is proper preparation of the structure. Dope-proofing, covering edges which are likely to wear the fabric, preparation of plywood surfaces and similar operations, will do much, if properly done, toward insuring a well-appearing and long-lasting job.

(i) DOPE-PROOFING. Treat all parts of the structure which come in contact with doped fabric with a protective coating such as aluminum foil, dope-proof paint or cellulose tape. Clad aluminum and stainless steel parts need not be dope-proofed.

(ii) CHAFE POINTS. All points of the structure such as sharp edges, bolt heads, etc., which are likely to chafe or wear the covering should be covered with doped-on fabric strips or covered with an adhesive tape. After the cover has been installed, the chafe points of the fabric should be reinforced by doping of fabric patches. Where a stronger reinforcement is required, a cotton duck or leather patch should be sewed to the fabric patch and then doped in place. All portions of the fabric pierced by wires, bolts or other projections should be reinforced.

(iii) INTER-RIB BRACING. Conventional wing ribs, which do not have permanent inter-rib bracing, should be tied in position by means of cotton tape running parallel to the beams. Apply the tape bracing to both the top and bottom capstrips, maintained parallel to the plane of the cover rather than diagonally between the top and bottom capstrips. Apply the tape continuously with one turn around successive capstrips, arranged so that the tape between the ribs is separated from the cover by a distance equal to the depth of the capstrip. Tie the turn of tape around each capstrip by means of a short length of lacing cord.

(iv) PREPARATION OF PLYWOOD SURFACES FOR COVERING. Prior to covering plywood surfaces with fabric, prepare the surface by cleaning and applying sealer and dope.

(*a*) CLEANING. Sand all surface areas which have been smeared with glue in order to expose a clean wood surface. Remove loose deposits such as wood chips and sawdust. Remove oil or grease spots by carefully washing with naphtha.

(*b*) APPLICATION OF SEALER AND DOPE. Apply one brush coat or two dip coats (wiped) of a dope-proof sealer such as Specification AN–TT–V–116 thinned to 30 percent non-volatile content and allow to dry 2 to 4 hours. Finally before covering, apply two brush coats of clear dope allowing the first coat of dope to dry approximately 45 minutes before applying the second coat.

(4) SEAMS.

(i) LOCATION OF SEAMS. Seams parallel to the line of flight are preferable; however, spanwise seams are acceptable.

(ii) SEWED SEAMS.

(*a*) Machine-Sewed Seams—Parts D, E, and F of Figure 2–1. Machine-sewed seams should be of the folded-fell or French-fell types. Where selvage edges or pinked edges are joined, a plain lap seam is satisfactory.

(*b*) Hand-sewing or tacking should begin at the point where machine-sewing stops and should continue to the point where machine-sewing or uncut fabric is again reached. Hand-sewing should be locked at intervals of 6 inches, and the seams should be properly finished with a lock stitch and a knot (Figure 2–5). At the point where the hand sewing or permanent tacking is necessary, the fabric should be so cut that it can be doubled under before sewing or permanent tacking is performed (Figure 2–1C). After hand-sewing has been completed, the temporary tacks should be removed. In hand-sewing there should be a minimum of four stitches per inch.

(*c*) A sewed spanwise seam on a metal or wood-covering leading edge should be covered with pinked-edge surface tape at least 4 inches wide.

(*d*) A sewed spanwise seam at the trailing edge should be covered with pinked-edge surface tape at least 1½ inches wide.

(*e*) Sewed spanwise seams on the upper or lower surface should be made in a manner that the amount of protuberance is a minimum. The seam should be covered with pinked edge tape at least 3 inches wide.

(*f*) Sewed seams parallel to the line of flight (chordwise) should not be placed over a rib or be so placed that the lacing will be through or across such a seam.

(iii) DOPED SEAMS.

(*a*) A lapped and doped spanwise seam on a metal or wood-covered leading edge should be covered with pinked edge surface tape at least 8 inches wide.

(*b*) A lapped and doped spanwise seam at the trailing edge should be covered with pinked edge surface tape at least 1½ inches wide.

(5) COVERING METHODS.

(i) THE ENVELOPE METHOD. The envelope method of covering is accomplished by sewing together widths of fabric cut to specified dimensions and machine sewn to form an envelope which can be drawn over the frame. The trailing and outer edges of the covering should be machine sewn unless the component is not favorably shaped for such sewing, in which case the fabric should be joined by hand sewing.

(ii) THE BLANKET METHOD. The blanket method of covering is accomplished by sewing together widths of fabrics of sufficient lengths to form a blanket over the surfaces of the frame. The trailing and outer edges of the covering should be joined by a plain over-throw or baseball stitch. For airplanes with placard never-exceed speed of 150 miles per hour or less, the blanket may be lapped at least 1 inch and doped to the frame or the blanket, lapped at least 4 inches at the nose of metal or wood-covered leading edges, doped, and finished with pinked-edge surface tape at least 8 inches wide. In fabricating both the envelope and blanket coverings, the fabric should be cut in lengths sufficient to pass completely around the frame, starting at the trailing edge and returning to the trailing edge.

(6) REINFORCING TAPE. Reinforcing tape of at least the width of the capstrips should be placed under all lacing. In the case of wings with plywood or metal leading-edge covering, the reinforcing tape need be brought only to the front spar on the upper and lower surfaces.

(i) USE OF ANTI-TEAR STRIPS. On aircraft with never-exceed speed in excess of 250 miles per hour, anti-tear strips are recommended under reinforcing tape on the upper surface of wings, and the bottom surface of that part of the wing in the slipstream. Where the anti-tear strip is used on both the top and bottom surfaces, pass it continuously up to and around the leading edges and back to the trailing edge. Where the strip is used only on the top surface, carry it up to and around the leading edge and back on the lower surface as far aft as the front beam. For this purpose the slipstream should be considered as being equal to the propeller diameter plus one extra rib space on each side. Cut anti-tear strips from the same material as used for covering and wide enough to extend beyond the reinforcing tape on each side so as to engage the lacing cord. Attach the strips by applying dope to that part of the fabric to be covered by the strip and applying dope freely over the strip.

(7) LACING.

(i) WING LACING. Both surfaces of fabric covering on wings and control surfaces should be securely fastened to the ribs by lacing cord or any other method originally approved for the aircraft. Care should be taken to insure that all sharp edges against which the lacing cord may bear are protected by tape in order to prevent abrasion of the cord. Separate lengths of lacing cord should be joined by the splice knot shown in Figure 2-4. The common square knot, which has a very low slippage resistance, should not be used for this purpose. The utmost care should be exercised to assure uniform tension and security of all stitches. The first or starting stitch should be made with a double loop by the method illustrated in Figure 2-6. All subsequent stitches should be made with a single loop and tied off with the standard knot for rib lacing (modified seine type), shown in Figure 2-5. The spacing between the starting stitch and the next stitch should be one-half the normal stitch spacing. All tie-off knots should be placed on the middle of the reinforcing tape on the bottom surface, or along the edge of the lower capstrip. The seine knot admits a possibility of improper tightening, resulting in a false (slip) form with greatly reduced efficiency and should not be used for stitch tie-offs. The tie-off knot for the last stitch should be locked by an additional half-hitch. Where stitching ends, as at the rear beam and at the trailing edge, the last two stitches should be spaced at one-half normal spacing. Under no circumstances should tie-off knots be pulled back through the lacing holes.

(ii) DOUBLE-LOOP LACING. The double-loop lacing illustrated in Figure 2-7 represents a method for obtaining higher strengths than possible with the standard single lacing. When using the double loop lacing, the tie-off knot should be made by the method shown in Figure 2-8.

(iii) FUSELAGE LACING. Fabric lacing is also necessary in the case of deep fuselages, and on fuselages where former strips and ribs shape the fabric to a curvature. In the latter case the fabric should be laced at intervals to the formers. The attachment of the fabric to fuselages should be so accomplished as to be at least the equivalent in strength and reliability to that used by the manufacturer of the airplane.

(8) STITCH SPACING. The stitch spacing should not exceed the spacing approved on the original aircraft. In case the spacing cannot be ascertained due to destruction of the covering, acceptable rib-stitch spacing may be found in Figure 2-2.

The lacing holes should be placed as near to the capstrip as possible in order to minimize the tendency of the cord to tear the fabric. All lacing cord should be lightly waxed with beeswax for protection. In case waxed braided cord is used, this procedure is unnecessary. (See Table 2-2 for acceptable lacing cords.)

(9) FINISHING TAPE. All lacing should be covered with tape of at least the quality and width as was used on the original airplane. This tape should not be applied until the first coat of dope has dried. All inspection openings should be reincorporated into the covering, and the fabric around them and along leading edges reinforced with tape. Where wear or friction is induced by moving parts or fittings, a leather patch should be sewed to a fabric patch and doped in place.

(10) SPECIAL FASTENERS. When repairs are made to fabric surfaces attached by special mechanical methods, the original type of fastening should be duplicated.

When self-tapping screws are used for the attachment of fabric to the rib structure, the following procedure should be observed:

(i) The holes should be redrilled where found necessary due to wear, distortion, etc., and in such cases a screw one size larger should be used as a replacement.

(ii) The length of the screw should be sufficient so that at least two threads of the grip (threaded part) extend beyond the rib capstrip.

(iii) A thin washer, preferably celluloid, should be used under the heads of screws and pinked-edge tape should be doped over each screw head.

(c) DOPING.

(1) THINNING OF DOPES. Dopes are generally supplied at a consistency ready for brush coats. For spraying operations practically all dopes require thinning. Thinning directions are usually listed on the container label. Where thinning operations are not supplied, thin the dope with a thinner made for the type of dope being used, until suitable brushing and/or spraying properties are obtained. The amount of thinner to be used will depend on the dope, atmospheric conditions, the spraying equipment, the spraying technique of the operator, and the type of thinner employed. The thinning of dopes influences the drying time and tautening properties of the finish and it is necessary that it be done properly. Determine the amount of thinner necessary by using experimental panels in order to ascertain the conditions which prevail locally at the time of application of the dope.

(2) BLUSHING AND USE OF BLUSH-RETARDING THINNER. Blushing of dopes is very common when doping is accomplished under humid conditions. This condition is caused by the rapid evaporation of thinners and solvents, which lowers the temperature on the surface, causing condensation of moisture and producing the white appearance known as blush. Blushing tendencies are also increased if strong currents of air flow over the surface when applying dopes or immediately thereafter.

A blushed finish has very little protective or tautening value. Where the relative humidity is such that only a small amount of blushing is encountered in doping, this blushing may be eliminated by thinning the dope with a blush-retarding thinner and slightly increasing the room temperature. If it is not possible to correct humidity conditions in the dope room, suspend doping operations until more favorable atmospheric conditions prevail. The use of large amount of blush-retarding thinner is not advisable because of the undesirable drying properties accompanying the use of this material.

(3) NUMBER OF COATS. The total number of coats of dope should not be less than that necessary to result in a taut and well-filled finish job. A guide for finishing fabric-covered aircraft follows:

(i) Two coats of clear dope, brushed on and sanded after the second coat.

(ii) One coat of clear dope, either brushed or sprayed, and sanded.

(iii) Two coats of aluminum pigmented dope, sanded after each coat.

(iv) Three coats of pigmented dope

(the color desired), sanded and rubbed to give a smooth glossy finish when completed.

(v) Precaution should be taken not to sand heavily over the center portion of pinked tape and over spars in order not to damage the rib-stitching cords and fabric.

(4) TECHNIQUE. Apply the first two coats of dope by brush and spread on the surface as uniformly as possible and thoroughly work into the fabric. Exercise care not to work the dope through the fabric so that an excessive film is formed on the reverse side. The first coat should produce a thorough and uniform wetting of the fabric. To do so, work the dope with the warp and the filler threads for three or four brush strokes and stroke away any excess material to avoid piling up or dripping. Apply succeeding brush coats with only sufficient brushing to spread the dope smoothly and evenly.

When doping fabric over plywood or metal-covered leading edges, care should be taken to insure that an adequate bond is obtained between the fabric and the leading edge. Especial care should be taken when using predoped fabric to use a thinned dope in order to obtain a good bond between the fabric and the leading edge.

(5) APPLYING SURFACE TAPE AND REINFORCING PATCHES. Apply surface tape and reinforcing patches with the second coat of dope. Apply surface tape over all rib lacing and over all sewed seams as well as at all other points of the structure where tape reinforcements are indicated.

(6) INSTALLATION OF DRAINAGE GROMMETS. With the second coat of dope, install drainage grommets on the underside of airfoils at the trailing edge and as close to the rib as practicable. On fuselages, install drainage grommets at the center of the underside in each fuselage bay, located so that the best possible drainage is effected. On seaplanes it is recommended to install special shielded grommets, sometimes called marine grommets, to prevent the entry of spray. Also use this type of grommet on landplanes in that part of the structure which is subject to splash from the landing gear when operating from wet and muddy fields. Plastic-type grommets are doped directly to the covering. Where brass grommets are used, mount them on fabric patches and then dope to the covering. After the doping scheme is completed, open the drain holes by cutting out the fabric with a small-bladed knife. Do not open drainage grommets by punching.

(7) COMMMON DOPE TROUBLES.

(i) In cold weather, dopes become quite viscous. Cold dopes pull and rope under the brush, and if thinned sufficiently to spray, lack body when dry. Prior to use, allow dopes to come to a temperature approximately that of the dope room—24° C. (75° F.).

(ii) Orange peel and pebble effect result from insufficiently thinned dope or when the spray gun is held too far from the surface being sprayed.

(iii) Runs, sags, laps, streaks, high and low spots are caused by improperly adjusted spraying equipment or improper spraying technique.

(iv) Pin holes may be caused by water or oil entering the spray gun. Drain air compressors, air regulators, and air lines daily.

(v) Wet areas on a doped surface indicate that oil, grease, soap, etc., had not been properly removed before doping.

(d) REPAIRS TO FABRIC COVERING. Repairs to fabric-covered surfaces should be made in a manner that will return the original strength and tautness to the fabric. Sewed repairs and unsewed (doped-on patches or panels) may be made.

(1) REPAIR OF TEARS IN FABRIC. Tears should be repaired as shown in Figure 2-3 by sewing the torn edges together using a baseball stitch and doping a piece of pinked-edge fabric over the tear. If the tear is a straight rip, the sewing is started at one end so that as the seam is made the edges will be drawn tightly together throughout its entire length. If the opening is V-shaped, as is often the case when openings are cut in wings to inspect the internal structure, the

sewing should start at the corner or point so that the edges of the cover will be held in place while the seams are being made. The sewing is done with a curved needle and well-waxed thread. Clean the surface to be covered by the patch by rubbing the surface with a rag dipped in dope and wiping dry with a clean rag, or by scraping the surface with a putty knife after it has been softened with fresh dope. Dope solvent or acetone may be used for the same purpose but care should be taken that it does not drop through on the inside of the opposite surface causing the dope to blister. A patch of sufficient size should be cut from airplane cloth to cover the tear and extend at least 1½ inches beyond the tear in all directions. The edges of the patch should either be pinked similar to surface tape or frayed out about ¼ inch on all edges.

(2) SEWED PATCH REPAIR. When the damage is such that it will not permit sewing the edges together, a sewed-in repair patch may be used if the damage is not longer than 16 inches in any one direction (see Figure 2–3). Cut out the damaged section making a round or oval-shaped opening trimmed to a smooth contour. Clean the area of the old fabric to be doped as indicated in CAM 18.20–2 (d) (1). Turn the edges of the patch ½ inch and sew to the edges of the opening. Before sewing, fasten the patch at several points with a few temporary stitches to facilitate sewing the seams. After the sewing is completed, clean the area of the old fabric to be doped as indicated for small repairs and then dope the patch in the regular manner. Apply surface tape over the seams with the second coat of dope. If the opening extends over or closer than 1 inch to a rib or other laced member, the patch should be cut to extend 3 inches beyond the member. After sewing has been completed, the patch should be laced to the rib over a new section of reinforcing tape using the methods of CAM 18.20–2 (b) (7) (i). The old rib lacing and reinforcing tape should not be removed.

(3) REPAIR BY SEWING IN REPAIR PANEL. When the damaged area exceeds 16 inches in any direction a new panel should be installed.

(i) Remove the surface tape from the ribs adjacent to the damaged area and from the trailing and leading edges of the section being repaired. Leave the old reinforcing tape in place.

(ii) Cut the old fabric along a line approximately 1 inch from the center of the ribs on the sides nearest to the injury, and continue the cuts to completely remove the damaged section. The old fabric should not be removed from the leading and trailing edge unless both upper and lower surfaces are being re-covered. Do not remove the reinforcing tape and lacing at the ribs.

(iii) Cut a patch to extend from the trailing edge up to and around the leading edge and back approximately to the front beam. The patch should extend approximately 3 inches beyond the ribs adjacent to the damage.

(iv) Clean the area of the old fabric to be covered by the patch, put the patch in place, stretch taut and pin. After the patch is pinned in place, fold under the trailing and leading edges of the patch ½ inch and sew to the old fabric. Fold the side edges under ½ inch and sew to the old cover. After completion of the sewing, place reinforcing tape over the ribs under moderate tension and lace down using the methods of CAM 18.20–2 (b) (7) (i). Remove the temporary pinning.

(v) Give the panel a coat of clear dope and allow to dry. Install surface tape with the second coat of dope, over the reinforcing tape and over the edges of the panel. Finish the doping scheme using regular doping procedures.

This type of repair may be extended to cover both the upper and lower surfaces and to cover several rib bays if necessary. The panel must be laced to all the ribs covered.

(4) UNSEWED (DOPED ON) REPAIRS. Unsewed (doped on) repairs may be made on all aircraft fabric-covered surfaces provided the newer-exceed speed is not greater than 150 miles per hour. A doped patch repair may be used if the damage does

not exceed 16 inches in any direction. Cut out the damaged section making a round or oval-shaped opening trimmed to a smooth contour. Clean the edges of the opening which are to be covered by the patch with grease solvent. If there is an excessively thick finish of old dope on the fabric, sand off or wash off the area around the patch with dope thinner. Support the fabric from underneath while sanding.

For holes up to 8 inches in size, make the fabric patch of sufficient size to provide a lap of at least 2 inches around the hole. On holes over 8 inches in size, make the overlap of the fabric around the hole at least one-fourth the hole diameter with a maximum limit of lap of 4 inches. If the hole extends over a rib or closer than the required overlap to a rib or other laced member, the patch should be extended at least 3 inches beyond the rib. In this case, after the edges of the patch have been doped in place and the dope has dried, the patch should be laced to the rib over a new section of reinforcing tape in the usual manner. The old rib lacing and reinforcing tape should not be removed. All patches should have pinked edges, or if smooth, finished with pinked-edge surface tape.

(5) REPAIR BY A DOPED-IN PANEL. When the damage exceeds 16 inches in any direction, make the repair by doping in a new panel.

(i) Remove the surface tape from the ribs adjacent to the damaged area and from the trailing and leading edges of the section being repaired. Leave the old reinforcing tape and lacing in place. Next cut the fabric along a line approximately 1 inch from the ribs on the sides nearest to the injury, and continue the cuts to completely remove the damaged section. The old fabric should not be removed from the leading and trailing edge unless both upper and lower surfaces are being re-covered.

(ii) Cut a patch to run around the trailing edge 1 inch and to extend from the trailing edge up to and around the leading edge and back approximately to the front beam. The patch should extend approximately 3 inches beyond the ribs adjacent to the damage.

As an alternative attachment on metal or wood-covered leading edges, the patch may be lapped over the old fabric at least 4 inches at the nose of the leading edge, doped, and finished with at least 8 inches of pinked-edge surface tape.

(iii) Clean the area of the old fabric that is to be covered by the patch and apply a generous coat of dope to this area. Put the new panel in place, pull as taut as possible, and apply a coat of dope to the portion of the panel which overlaps the old fabric. After this coat has dried, apply a second coat of dope to the over-lapped area and let dry.

(iv) Place reinforcing tape over the ribs under moderate tension and lace down in the approved manner.

(v) Give the panel a coat of clear dope and allow to dry. Install surface tape with the second coat of dope over the reinforcing tape and over the edges of the panel. Finish the doping scheme using the regular doping procedure.

This type of repair may be extended to cover both the upper and lower surfaces and to cover several rib bays if necessary. The panel should be laced to all the ribs covered, and doped or sewed as in the blanket method.

(e) TESTING OF FABRIC COVERING. Tensile testing of fabric is a practical means for determining whether a fabric covering has deteriorated to a point where re-covering is necessary. The testing may be carried out in accordance with the procedures set forth in existing industry or Government specifications such as Federal Specification CCC-T-191a, American Society for Testing Materials D39-39, and others. In all cases the specimens should be tested in the undoped condition. The use of acetone or dope thinner is suggested as a means of removing the dope.

(1) STRENGTH CRITERIA FOR AIRCRAFT FABRIC.

(i) Present minimum strength values for new aircraft fabric covering are contained in Table 2-1.

(ii) The maximum permissible deterioration for used aircraft fabric based on a large number of tests is 30 percent. Fabric which has less than 70 percent of the original required tensile strength should not be considered airworthy. Table 2-1 contains the minimum tensile strength values for deteriorated fabric as tested in the undoped condition.

(iii) In cases where light aircraft operators use the Grade "A" type fabric, but are only required to use "Intermediate" Grade fabric, the Grade "A" material is still considered airworthy, provided it has not deteriorated, as tested in the undoped condition, below 46 pounds, i. e., 70 percent of the originally required tensile strength value for new "Intermediate" fabric.

18.20-3 METAL AIRCRAFT STRUCTURES. ***(CAA policies which apply to section 18.20.)***

(a) METAL CONSTRUCTION.

(1) IDENTIFICATION AND INSPECTION OF MATERIALS. Identification and inspection of materials should be conducted in accordance with CAM 18.20-6.

(2) CORROSION PREVENTION TREATMENT, CLEANERS, AND PAINT REMOVERS. Corrosion prevention treatment, cleaning and paint removing should be accomplished in accordance with CAM 18.20-7.

(3) BOLTS, SCREWS, AND FASTENERS. Acceptable means of attachment are listed in CAM 18.20-5.

(4) FLUTTER PRECAUTIONS. When repairing control surfaces, especially on high-performance airplanes, care should be exercised that the repairs do not involve the addition of weight aft of the hinge line. Such procedure may adversely disturb the dynamic and static balance of the surface to a degree which would induce flutter. As a general rule it will be necessary to repair control surfaces in such a manner that the structure is identical to the original so that the weight distribution is not affected in any way.

(5) BRAZING. Brazing may be used for major repairs in aircraft only when the particular application was originally approved for the aircraft. Brazing is not generally suitable for repair of welds in steel structures due to lower strength values of the brazed joint as compared to welded joints.

Due to the large number of brazing alloys used, it is difficult to be certain that the one used for repairing a brazed joint would not combine with the original brazing alloy to produce a low strength joint.

In cases where it is necessary to re-apply copper alloy brazing material on a steel surface more than once, and particularly if temperatures over 2,000° F. are reached, there is a possibility that brazing metal may penetrate between the grains in the steel to an extent that may cause cracking.

Copper brazing of steel is normally done in a special furnace having a reducing atmosphere, and at a temperature so high that field repairs are seldom feasible. If copper brazing is attempted without a controlled atmosphere, the copper will probably not completely wet and fill the joint.

(i) NONSTRUCTURAL REPAIRS. Nonstructural repairs may be made by brazing.

(b) WELDED STEEL STRUCTURES.

(1) GENERAL. Oxyacetylene or electric arc welding may be utilized for repair of aircraft structural elements. Most aircraft structures are fabricated from one of the weldable alloys; however, careful consideration should be given to the alloy being welded since all alloys are not readily weldable. In general, the more responsive an alloy steel is to heat treatment, the less suitable it is for welding, because the greater will be its tendency to become brittle and lose its ductility in the welded area. The following steels are readily weldable: plain carbon, nickel steels of the S.A.E. 2300 series, chrome-nickel alloys of the S.A.E. 3100 series, chrome-molybdenum steels of the S.A.E. 4100 series and low nickel-chrome-molybdenum steel of the S.A.E. 8600 series.

(i) PREPARATION FOR WELDING. The elements to be welded should be properly held in place by welding jigs or fixtures which are sufficiently rigid to prevent misalignment due to expansion and contraction of the heated material and which positively and accurately locate the relative positions of the pieces to be welded together.

(ii) CLEANING PRIOR TO WELDING. The parts to be welded should always be cleaned by wire brushing or other similar methods. When a wire brush is used, care should be taken never to use a brush of dissimilar metal, for example, brass or bronze. The small deposit left by a brass or bronze brush will materially weaken the weld and may cause cracking and subsequent failure of the weld. In case members were metallized, the surface metal may be removed by careful sandblasting.

(iii) CONDITION OF COMPLETED WELD. The finished weld should incorporate the following characteristics:

(*a*) The seam should be smooth and of uniform thickness.

(*b*) The weld metal should taper off smoothly into the base metal.

(*c*) No oxide should be formed on the base metal at a distance of more than ½ inch from the weld.

(*d*) The weld should show no signs of blow holes, porosity or projecting globules.

(*e*) The base metal should show no signs of pitting, burning, cracking, or distortion.

(*f*) The depth of penetration should be sufficient to insure fusion of base metal and filler rod.

(*g*) Welding scale should be removed by wire brushing or sandblasting.

(iv) PRACTICES TO GUARD AGAINST. No welds should be filed in an effort to make a smooth appearing job, as such treatment causes a loss in strength. Welds should not be filled with solder, brazing metal, or any other filler. When it is necessary to reweld a joint which was previously welded, all old-weld material should be thoroughly removed before rewelding. Never weld over a weld if it can be avoided because continual heating causes the material to lose its strength and to become brittle. Never weld a joint which has been previously brazed.

(v) TORCH SIZE (OXYACETYLENE WELDING). The torch tips should be of proper size for the thickness of the material to be worked on. The commonly used sizes which experience has proven to be satisfactory are:

Thickness of steel in inches	Diameter of hole in tip	Drill Size
0.015 to 0.031	0.026	71
0.031 to 0.065	.031	68
0.065 to 0.125	.037	63
0.125 to 0.188	.042	58
0.188 to 0.250	.055	54
0.250 to 0.375	.067	51

(vi) WELDING RODS AND ELECTRODES. Welding rods and electrodes for various applications have special properties suitable for the application intended. Table 3–1 outlines oxyacetylene welding rod properties, and Table 3–2 outlines arc welding electrode properties.

(vii) ROSETTE WELDS. Rosette welds are generally employed to fuse an inner reinforcing tube (liner) with the outer member. Where a rosette weld is used, the hole should be made in the outside tube only and be of a sufficient size to insure fusion of the inner tube. A hole diameter of approximately one-fourth the tube diameter of the outer tube has been found to serve adequately for this purpose. In cases of tightly fitting sleeves or inner liners, the rosettes may be omitted.

(viii) HEAT-TREATED MEMBERS. Members which depend on heat-treatment for their original physical properties should be welded using a welding rod suitable for producing heat-treated values comparable to those of the original member (see CAM 18.20–3 (b) (1) (vi)). Such members must be reheat-treated to the manufacturer's specifications after welding.

(ix) STEEL PARTS NOT TO BE WELDED.

(*a*) BRACE WIRES AND CABLES. Airplane parts that depend for their proper functioning on strength properties developed by cold working should not be welded. In this classification are streamlined wires and cables.

(*b*) BRAZED AND SOLDERED PARTS. Brazed or soldered parts must not be welded, as the brazing mixture or solder will penetrate the hot steel and weaken it.

(*c*) ALLOY STANDARD PARTS. Alloy steel parts which have been heat-treated to improve their physical properties should not be welded. This pertains particularly to aircraft bolts, turnbuckle ends, axles and other heat-treated alloy steel parts.

(2) REPAIR OF TUBULAR MEMBERS.

(i) INSPECTION. Prior to repairing tubular members, the structure surrounding any visible damage should be carefully examined to insure that no secondary damage remains undetected. Secondary damage may be produced in some structure remote from the location of the primary damage by the transmission of the damaging load along the tube. Damage of this nature usually occurs where the most abrupt change in direction of load travel is experienced. If this damage remains undetected, loads applied in the normal course of operation may cause failure of the part. Visually examine closely all joints for cracks, welding flaws or other defects.

(*a*) LOCATION AND ALIGNMENT OF WELDS. Unless otherwise noted welded steel tubing may be spliced or repaired at any joint along the length of the tube including the middle fourth. Particular attention should be paid to proper fit and alignment to avoid eccentricities.

(ii) MEMBERS DENTED AT A CLUSTER. Dents at a steel-tube cluster joint may be repaired by welding an especially formed steel patch plate over the dented area and surrounding tubes, as shown in Figure 3–1. To prepare the patch plate, cut a section of steel sheet of the same material and thickness as the heaviest tube damaged. Trim the reinforcing plate so that the fingers extend over the tubes a minimum of 1.5 times the respective tube diameter as shown in the figure. Remove all the existing finish on the damaged cluster joint area to be covered by the reinforcing plate. The reinforcing plate may be formed before any welding is attempted, or it may be cut and tack-welded to one or more of the tubes in the cluster joint, then heated and formed around the joint to produce a smooth contour. Apply sufficient heat to the plate while forming so that there is generally a gap of no more than 1/16 inch from the contour of the joint to the plate. In this operation avoid unnecessary heating and exercise care to prevent damage at the apex of the angle formed by any two adjacent fingers of the plate. After the plate is formed and tack-welded to the cluster joint, weld all the plate edges to the cluster joint.

(iii) MEMBERS DENTED IN A BAY. Dented, bent, cracked or otherwise damaged tubular members may be repaired by using a split sleeve reinforcement, after first carefully straightening the damaged member, and in the case of cracks, drilling No. 40 (.098) stop holes at the ends of the crack.

(*a*) REPAIR BY WELDED SLEEVE. This repair is outlined in Figure 3–2. Select a length of steel tube sleeve having an inside diameter approximately equal to the outside diameter of the damaged tube and of the same material and at least the same wall thickness. Diagonally cut the sleeve reinforcement at a 30° angle on both ends so that the minimum distance of the sleeve from the edge of the crack or dent is not less than 1½ times the diameter of the damaged tube. Cut through the entire length of the reinforcing sleeve and separate the half sections of the sleeve. Clamp the two sleeve sections to the proper positions on the affected areas of the original tube. Weld the reinforcing sleeve along the length of the two sides, and weld both ends of the sleeve to the damaged tube as shown in the figure. The filling of dents or cracks with welding rod in place of reinforcing the member is not acceptable.

(*b*) REPAIR BY BOLTED SLEEVE. Due to the large percentages of tube areas removed by the bolt holes, bolted sleeve repairs should not be used on welded steel structures, without prior approval of the repair by the Civil Aeronautics Administration.

(iv) WELDED-PATCH REPAIR. Dents or holes in tubing may be repaired by a welded patch of the same material and one gage thicker, as shown in Figure 3-3 provided:

(*a*) DENTED TUBING.

(*1*) Dents are not deeper than $\frac{1}{10}$ of tube diameter, do not involve more than $\frac{1}{4}$ of the tube circumference, and are not longer than tube diameter.

(*2*) Dents are free from cracks, abrasions and sharp corners.

(*3*) The dented tubing can be substantially re-formed without cracking before application of the patch.

(*b*) PUNCTURED TUBING. Holes are not longer than tube diameter and involve not more than $\frac{1}{4}$ of tube circumference.

(*c*) LOCATION OF PATCH. No part of the patch is permitted in the middle third of the tube. The patch should not overlap a tube joint.

(v) SPLICING BY INNER SLEEVE METHOD. If the damage to a structural tube is such that partial replacement of the tube is necessary, the inner sleeve splice shown in Figure 3-4 is recommended, especially where a smooth tube surface is desired. Diagonally cut out the damaged portion of the tube, and remove the burr from the edges of the cuts by filing or similar means. Diagonally cut a replacement steel tube of the same material and diameter and at least the same wall thickness to match the length of the removed portion of the damaged tube. At each end of the replacement tube allow a $\frac{1}{8}$-inch gap from the diagonal cuts to the stubs of the original tube. Select a length of steel tubing of the same material and at least the same wall thickness and of an outside diameter approximately equal to the inside diameter of the damaged tube. This inner sleeve tube material should fit snugly within the original tube, with a maximum diameter difference of $\frac{1}{16}$ inch. From this inner sleeve tube material cut two sections of tubing, each of such a length that the ends of the inner sleeve will be a minimum distance of $1\frac{1}{2}$ tube diameters from the nearest end of the diagonal cut.

If the inner sleeve fits very tightly in the replacement tube, chill the sleeve with dry ice or in cold water. If this is insufficient, polish down the diameter of the sleeve with emery cloth. Weld the inner sleeve to the tube stubs through the $\frac{1}{8}$-inch gap between the stubs completely filling the $\frac{1}{8}$-inch gap forming a weld bead over the gap.

(vi) SPLICING BY OUTER SLEEVE METHOD. If partial replacement of a tube is necessary, an outer sleeve splice using a replacement tube of the same diameter may be made. However, the outer sleeve splice requires the greatest amount of welding and, therefore, it should be used only where the other splicing methods are not suitable. Information on the replacement by use of the welded outside sleeve method is given in Figures 3-5 and 3-6.

Squarely cut out the damaged section of the tube. Cut a replacement steel tube of the same material and diameter and at least the same wall thickness to match the length of the removed portion of the damaged tube. This replacement tube must bear against the stubs of the original tube with a total tolerance not to exceed $\frac{1}{32}$ inch. Select a length of steel tubing of an inside diameter approximately equal to the outside diameter of the damaged tube and of the same material and at least the same wall thickness. This outer sleeve tube material should fit snugly about the original tube with a maximum diameter difference of $\frac{1}{16}$ inch. From this outer sleeve tube material, cut two sections of tubing diagonally or fishmouth, each of such a length that the nearest ends of the outer sleeve are a minimum distance of $1\frac{1}{2}$ tube diameters from the ends of the cut on the original tube. Use a fishmouth-cut sleeve wherever possible. Remove the burr from all the edges of the sleeves, replacement tube, and original tube stubs. Slip the two sleeves over the replacement tube, line up the replacement tube with the original tube stubs, and slip the sleeves out over the center of each joint. Adjust the sleeves to suit the area and to provide maximum reinforcement. Tack-weld the two sleeves to the replacement tube in two places before welding. Apply a uniform weld around both ends of one of the reinforcing sleeves and allow the weld to cool. Then weld around both ends of the remaining reinforcing tube. Allow one sleeve weld to cool before welding the remaining tube, to prevent undue warping.

(vii) SPLICING USING LARGER DIAMETER REPLACEMENT TUBES. This method of splicing structural tubes shown in Figure 3–7 requires the least amount of cutting and welding. However, this splicing method cannot be used where the damaged tube is cut too near the adjacent cluster joints or where bracket mounting provisions make it necessary to maintain the same replacement tube diameter as the original. As an aid in installing the replacement tube, squarely cut the original damaged tube, leaving a minimum short stub equal to 2½ tube diameters on one end and a minimum long stub equal to 4½ tube diameters on the other end.

Select a spare length of steel tube of the same material and at least the same wall thickness, having an inside diameter approximately equal to the outside diameter of the damaged tube. This replacement tube material should fit snugly about the original tube with a maximum diameter difference of $\frac{1}{16}$ inch. From this replacement tube material, diagonally or fishmouth, cut a section of tubing of such a length that each end of the tube is a minimum distance of 1½ tube diameters from the end of the cut on the original tube. Use a fishmouth-cut replacement tube wherever possible. However, a diagonally cut tube may also be used. Remove the burr from the edges of the replacement tube and the original tube stubs. If a fishmouth cut is used, file out the sharp radius of the cut with a small, round file. Spring the long stub of the original tube from the normal position; slip the replacement tube over the long stub, then back over the short stub. Center the replacement tube between the stubs of the original tube. In several places tack-weld one end of the replacement tube; then weld completely around the end. In order to prevent distortion, allow the weld to cool completely; then weld the remaining end of the replacement tube to the original tube.

(3) REPAIRS AT BUILT-IN FUSELAGE FITTINGS. Repairs of built-in fuselage fittings may be accomplished in a manner as shown in Figure 3–8. Splices should be made in accordance with the methods described in the foregoing sections. The following sections outline the different methods as shown in the figure.

(i) TUBE OF LARGER DIAMETER THAN ORIGINAL. A tube (sleeve) of larger diameter than original is used in the method shown in Figure 3–9. This necessitates reaming the fitting holes (at longeron) to a larger diameter. The sleeve should extend approximately 6 inches forward (left of fitting) of the joint and 8 inches aft right of fitting). The forward splice should be a 30° scarf splice. The rear longeron (right) should be cut off approximately 4 inches from the centerline of the joint and a spacer 1 inch long fitted over the longeron. This spacer and longeron should be edge welded. A tapered V-cut approximately 2 inches long should then be made in the aft end of the outer sleeve. The end of the outer sleeve should be swaged to fit the longeron and welded.

(ii) TUBE OF SAME DIAMETER AS ORIGINAL. In this method, shown in Figure 3–9 the new section of tube is the same size as the longeron forward (left) of the fitting. The rear end (right) of the tube is cut at 30° and forms the outside sleeve of a scarf splice. A sleeve is centered over the forward joint as indicated.

(iii) SIMPLE SLEEVE. The longeron is assumed the same size on each side of the fitting in this case, in Figure 3–9, and is repaired by a simple sleeve of larger diameter than the longeron.

(iv) LARGE DIFFERENCE IN LONGERON DIAMETER EACH SIDE OF FITTING. Figure 3–9 (D) assumes that there is a quarter of an inch difference in the diameter of the longeron on the two sides of the fitting. The section of longeron forward (left) of the fitting is cut at 30° and a section of tubing of the same size as this tube and of such length as to extend well to the rear (right) of the fitting is slipped through it. One end is cut at 30° to fit the 30° scarf at left and the other end fishmouthed as shown. This makes it possible to insert a tube of such diameter as to form an inside sleeve for the tube on the left of the fitting and an outside sleeve for the tube on the right of the fitting.

(4) ENGINE MOUNTS.

(i) GENERAL. All welding on an engine mount should be of the highest quality, since vibration tends to accentuate any minor defect present. Engine mount members should preferably be repaired by using a larger diameter replacement tube telescoped over the stub of the original member, and using fishmouth and rosette welds. However, 30° scarf welds in place of the fishmouth welds will be considered acceptable for engine mount repair work.

(ii) CHECK OF ALINEMENT. Repairs to engine mounts should be governed by accurate means of checking alinement. When new tubes are used to replace bent or damaged ones, the original alinement of the structure must be maintained. This can be done by measuring the distance between points of corresponding members that have not been distorted, and by reference to the manufacturer's drawings.

(iii) CAUSE FOR REJECTION. If all members are out of alinement, the engine mount should be replaced by one supplied by the manufacturer or one which has been built from the manufacturer's drawings. The method of checking the alinement of the fuselage or nacelle points should be requested from the manufacturer.

(iv) ENGINE MOUNT RING DAMAGE. Minor damage such as a crack adjacent to an engine attachment lug may be repaired by rewelding the ring and extending a gusset or a mounting lug past the damaged area. Engine mount rings which have been extensively damaged should not be repaired but should be replaced unless the method of repair is specifically approved by an authorized representative of the Civil Aeronautics Administration.

(5) LANDING GEARS.

(i) ROUND TUBE CONSTRUCTION. Landing gears made of round tubing may be repaired using standard repairs and splices, as shown in Figures 3-2 and 3-8.

(ii) STREAMLINE TUBE CONSTRUCTION. Landing gears made of streamlined tubing may be repaired by any one of the methods shown in Figures 3-9 and 3-12.

(iii) AXLE ASSEMBLIES. Representative types of repairable and nonrepairable landing gear axle assemblies are shown in Figure 3-13. The types as shown in A, B, and C of this figure are formed from steel tubing and may be repaired by any standard method shown in the preceding figures of this manual. However, it will always be necessary to ascertain whether or not the members are heat-treated.

The axle assembly as shown in Figure 3-13 D is, in general, of a nonrepairable type for the following reasons:

(*a*) The axle stub is usually made from a highly heat-treated nickel alloy steel and carefully machined to close tolerances. These stubs are usually replaceable and should be replaced if damaged.

(*b*) The oleo portion of the structure is generally heat-treated after welding and is perfectly machined to assure proper functioning of the shock absorber. These parts would be distorted by welding after machining.

(iv) SKI PEDESTALS. Damaged pedestals made of steel tubing may be repaired by using standard tube splices as shown in Figures 3-2 through 3-12.

(6) BUILT-UP TUBULAR WING OR TAIL SURFACE SPARS. Built-up tubular wing or tail surface spars may be repaired by using any of the standard splices and methods of repair shown in the figures of this manual provided the spars are not heat-treated. In the latter case the entire spar assembly would have to be reheat-treated to the manufacturer's specifications after com-

pletion of the repair. In general, this will be found less practicable than replacing the spar with one furnished by the manufacturer of the aircraft.

(7) WING AND TAIL SURFACE BRACE STRUTS. In general it will be found advantageous to replace damaged wing brace struts made either from round or streamlined tubing by new members purchased from the original manufacturer. However, there is no objection from an airworthiness point of view to repairing such members in a proper manner. An acceptable method in case streamlined tubing is used will be found in Figure 3-10. Similar members made of round tubes may be repaired using a standard splice, as shown in Figures 3-2, 3-4, or 3-5.

(i) LOCATION OF SPLICES. Steel brace struts may be spliced at any point along the length of the strut provided the splice does not overlap any part of an end fitting. The jury strut attachment is not considered an end fitting; therefore, a splice may be made at this point. The repair procedure and workmanship should be such as to minimize distortion due to welding and the necessity for subsequent straightening operations. Every repaired strut should be carefully observed during initial flights to ascertain that the vibration characteristics of the strut and attaching components have not been adversely affected by the repair. The check should cover a wide range of speed and engine power combinations.

(ii) FIT AND ALINEMENT. When making repairs to wing and tail surface brace members, particular attention should be paid to proper fit and alinement to avoid eccentricities.

(8) REPAIRS TO WELDED PARTS. Repairs to welded assemblies may be made by either of the following methods:

(i) REPLACING WELDED JOINT. Cutting out the welded joint and replacing it with one properly gusseted.

(ii) REPLACING WELD DEPOSIT. Chipping out the metal deposited by the weld process and rewelding after properly reinforcing the joint by means of inserts or external gussets.

(c) STAINLESS STEEL STRUCTURES.

(1) GENERAL. Structural components made from stainless steel, particularly the "18-8" variety (18 percent chrome, 8 percent nickel), joined by the very rapid process of spot welding, should be repaired only at the factory of origin or by a repair station designated by the manufacturer and rated by the Civil Aeronautics Administration to perform this type of work, unless the repair method incorporates bolted or riveted connections which are specifically approved by an authorized representative of the Civil Aeronautics Administration.

(2) SECONDARY STRUCTURAL AND NONSTRUCTURAL ELEMENTS. Elements such as tip bows or leading and trailing edge tip strips of wing and control surfaces may be repaired by soldering with a 50-50 lead-tin solder or a 60-40 alloy of these metals. For best results a flux of phosphoric acid (syrup) should be used. Since the purpose of a flux is to attack the metal so that the soldering will be effective, any excess flux should be removed by washing the joint. Due to the high heat conductivity of stainless steel, a soldering iron large enough to do the work properly must be used. Leaky spot welded seams in boat hulls, fuel tanks, etc., should be repaired in a similar manner.

(d) RIVETED OR BOLTED STEEL TRUSS TYPE STRUCTURES. Repairs to riveted or bolted steel truss type structures should be made employing the general principles outlined in the following sections on aluminum alloy structures. Methods of repair of vital members should specifically be approved by a representative of the Civil Aeronautics Administration.

(e) ALUMINUM AND ALUMINUM ALLOY STRUCTURES.

(1) GENERAL. Extensive repairs to damaged stressed skin on monocoque types of aluminum alloy structures should be made at the factory of origin or by a repair sta-

tion rated for this type of work. In any event such work should be undertaken only by a certificated mechanic thoroughly experienced in this type of work. The repairs should preferably be made in accordance with specific recommendations of the manufacturer of the aircraft. In many cases repair parts, joints, or reinforcements can be designed and proof of adequate strength shown, without the calculation of the actual loads and stresses, by properly considering the material and dimensions of the original parts and the riveted attachments. Examples illustrating the principles of this method as applied to typical repairs are given in this manual or may be found in textbooks on metal structures. An important point to bear in mind in making repairs on monocoque structures is that a repaired part must be as strong as the original with respect to all types of loads and general rigidity.

(i) USE OF ANNEALED ALLOYS FOR STRUCTURAL PARTS. The use of annealed 17S or 24S alloys for any structural repair of an aircraft where corrosion is likely to occur will not be considered satisfactory on account of poor corrosion resisting properties.

(ii) HYGROSCOPIC MATERIALS IMPROPERLY MOISTURE-PROOFED. The use of hygroscopic materials improperly moisture-proofed such as impregnated fabrics, leather and the like, in attempting to effect watertightness of joints and seams will not be considered acceptable practice.

(iii) DRILLING OVERSIZE HOLES. Great care should be exercised to avoid drilling oversize holes or otherwise decreasing the effective tensile area of wing spar capstrips, wing, fuselage, or fin longitudinal stringers, or other highly stressed tensile members. All repairs or reinforcements to such members should be done in accordance with factory recommendations or with the specific approval of a representative of the Civil Aeronautics Administration.

(iv) DISASSEMBLY PRIOR TO REPAIRING. If the parts to be removed are essential to the rigidity of the complete structure, the remaining structure should be adequately supported prior to disassembly, in such a manner as to prevent distortion and permanent damage to the remainder of the structure. Rivets may be removed by using special tools developed for the purpose or by center-punching the heads, drilling not quite through with a drill of the same size as the rivets, and shearing the heads off by a sharp blow with a small cold chisel. Rivet joints adjacent to the damaged parts should be inspected for partial failure (slippage) by removing one or more rivets to see if the holes are elongated or the rivets have started to shear.

(2) SELECTION OF MATERIAL FOR REPLACEMENT PARTS. In selecting the alloy, it is usually satisfactory to use 24S in place of 17S since the former is stronger. Hence, it will not be permissible to replace 24S by 17S unless the deficiency in strength of the latter material has been compensated by an increase in material thickness or the structural strength has been substantiated by tests or analyses. Information on the comparative strength properties of these alloys was well as 14S, R-301, 61S, 75S, etc., is contained in ANC-5a, "Strength of Metal Aircraft Elements." The choice of temper depends, of course, upon the severity of the subsequent forming operations. Parts having single curvature and straight bend lines with a large bend radius may be advantageously formed from heat-treated material, while a part such as a fuselage frame would have to be formed from soft annealed sheet and heat-treated after forming. Sheet metal parts which are to be left unpainted should be made of clad (aluminum coated) material. All sheet material and finished parts should be free from cracks, scratches, kinks, tool marks, corrosion pits, and other defects conducive to cracking.

(i) FORMING SHEET METAL PARTS. Bend lines should preferably be made to lie at an angle to the grain of the metal (preferably 90°). Before bending, all rough edges should be smoothed, burrs removed, and relief holes drilled at the ends of bend lines and at corners to prevent cracks from starting. For material in the heat-treated condition, the bend radius should be large. See Table 3-3 for recommended bend radii.

(3) HEAT TREATMENT.

(i) GENERAL. All structural aluminum alloy parts should be heat treated in accordance with the heat treatment instructions issued by the manufacturers of the materials. If the heat treatment produces warping, the parts should be straightened immediately after quenching. Parts riveted together should be heat treated before riveting, since heat treating after riveting causes warping. When riveted assemblies are heated in a salt bath, the salt cannot be entirely washed out of the crevices and causes corrosion.

(ii) QUENCHING IN HOT WATER OR AIR. The quenching of 17S or 24S alloys in water above 100° F, or air at any temperature after heat treatment will not be satisfactory. For clad material, when the use of cold water will result in too great a distortion of the finished part, the use of oil, hot water, water spray or forced air draft is satisfactory, provided the parts will not be subject to severe corrosion in service. Quenching in still air is not satisfactory.

(iii) TRANSFERING TOO SLOWLY FROM HEAT TREATMENT MEDIUM TO QUENCH TANK. Insufficiently rapid transfer of 17S or 24S alloys from the heat treatment medium to the quench tank will not be considered good practice. (An elapsed time of 10 to 15 seconds will, in many cases, result in noticeably impaired corrosion resistance).

(iv) REHEATING AT TEMPERATURES ABOVE BOILING WATER. Reheating at temperatures above that of boiling water of 17S or 24S alloys after heat treatment, and the baking of primers at temperatures above that of boiling water, will not be considered acceptable without subsequent complete and correct heat treatment, as such practice tends to impair the original heat treatment.

(4) RIVETING.

(i) IDENTIFICATION OF RIVET MATERIAL. Identification of rivet material is contained in CAM 18.20-5.

(ii) REPLACEMENT OF ALUMINUM ALLOY RIVETS. All protruding heat rivets (round-head, flat-head, and brazier-head) may be replaced by rivets of the same type or by AN-470 Universal-head rivets. Flush-head rivets should be used to replace flush-head rivets.

(*a*) REPLACEMENT RIVET SIZE AND STRENGTH. Replacements should be made with rivets of the same size and strength wherever possible. If the rivet hole has become enlarged, deformed, or otherwise damaged, the hole should be drilled or reamed for the next larger size rivet, care being taken, however, that the edge distances and spacings are not less than minimums listed in CAM 18.20-3 (e) (4) (ii) (*b*). Rivets may not be replaced by a kind of lower strength properties, unless the lower strength is adequately compensated for by an increase in size or a greater number of rivets, provided the edge distances and spacings are not less than the minimums listed in CAM 18.20-3(e) (4) (ii) (*b*).

(*b*) REPLACEMENT RIVET EDGE DISTANCES AND SPACINGS FOR SHEET JOINTS. Rivet edge distance is defined as the distance from the center of the rivet hole to the nearest edge of the sheet. Rivet spacing is the distance from the center of the rivet hole to the center of the adjacent rivet hole. Edge distances and spacings should not be less than the following:

(*1*) SINGLE ROW. Edge distances not less than two times the diameter of the rivet and spacing not less than three times the diameter of the rivet.

(*2*) DOUBLE ROW. Edge distance and spacing not less than the minimums shown in Figure 3-14.

(*3*) TRIPLE OR MULTIPLE ROWS. Edge distance and spacing not less than the minimums shown in Figure 3-14.

(iii) USE OF A17S-T3 ALUMINUM ALLOY REPLACEMENT RIVETS. It will be considered acceptable to replace all 17S-T3 rivets of $\frac{3}{16}$-inch diameter or less, and also all 24S-T4 rivets of $\frac{5}{32}$-inch diameter or less with A17S–T3 rivets for general repairs, provided the replacement rivets are $\frac{1}{32}$ inch greater in diameter than the rivets they replace, and provided the edge distances and spacings are not less than the minimums listed in CAM 18.20–3(e)(4)(ii)(*b*).

(iv) DRIVING OF RIVETS. A17S rivets may be driven in the condition received, but 17S rivets above $\frac{3}{16}$ inch in diameter and all 24S rivets should either be kept refrigerated in the "as quenched" condition until driven or be reheat-treated just prior to driving as they would otherwise be too hard for satisfactory riveting. Dimensions for formed flat rivet heads are shown in Figure 3–15, together with commonly found rivet imperfections, which should be guarded against.

(v) BLIND-TYPE AND HOLLOW RIVETS. Hollow rivets should not be substituted for solid rivets in load carrying members without specific approval of the application by a representative of the Civil Aeronautics Administration.

Blind rivets may be used in blind locations in accordance with the conditions listed in CAM 18.20–5, provided the edge distances and spacings are not less than the minimums listed in CAM 18.20–3(e)(4)(ii)(*b*).

(vi) NEW AND REVISED RIVET PATTERNS. A new or revised rivet pattern should be designed for the strength required in accordance with the specific instructions in CAM 18.20–3(e)(5)(vi) and 18.20–3(e)(5)(viii)(*d*).

A general rule for the diameter of rivets used to join dural sheets is to use a diameter approximately three times the thickness of the sheet, or somewhat larger for thin sheet. Rivets should not be used where they would be placed in tension tending to pull the heads off. A lap joint of thin sheets should be "backed up" by a stiffening section.

(5) REPAIR METHODS.

(i) PRECAUTIONS. When adding or replacing rivets adjacent or near to 17S or 24S rivets which have been installed previously, great care should be exercised or the older rivets will be loosened or may fail due to sharp vibrations in the structures caused by the action of the rivet gun and bucking bar. In every case all adjacent rivets should be carefully examined after the repair or alteration is finished to ascertain that they have not been harmed by adjacent operations.

Rivet holes should be drilled, round, straight, and free from cracks. The snap used in driving the rivets should be cupped slightly flatter than the rivet heads shown in Figure 3–15. Rivets should be driven straight and tight, but not overdriven or driven while too hard, since the finished rivet must be free from cracks. Information on special methods of riveting, such as flush riveting, may usually be obtained from manufacturer's service manuals.

(ii) SPLICING OF TUBES. Round or streamlined tubular members may be repaired by splicing as shown in Figure 3–16. Splices in struts should not overlap the fittings.

When solid rivets go completely through hollow tubes, their diameter should be at least one-eighth of the outside diameter of the outer tube. Rivets which are loaded in shear should be hammered only enough to form a small head, and no attempt should be made to form the standard round head. The amount of hammering required to form the standard round head often causes the rivet to buckle inside the tube. Satisfactory rivet heads may be produced in such installations by spinning, if the proper equipment is available. Correct and incorrect examples of this type of rivet application are incorporated in Figure 3–16.

(iii) REPAIRS TO 24S–T36 AND 75S–T6 ALLOY MEMBERS. Repairs involving 24S–T36 alloy members should be made with the same material. The 75S alloy has greater tensile strength than other commonly used aluminum alloys such as 14S and 24S and it is subject to somewhat greater notch sensitivity. In order to take advantage of its higher strength characteristics, particular attention should be paid in design of parts to avoid notches, small radii, large or rapid changes in cross-sectional area. In fabrication, more care should be taken to avoid processing and handling defects, such as machine marks, nicks, dents, burrs, scratches and forming cracks. Cold straightening or forming of 75S–T6 can cause cracking; hence, it may be advisable to limit this processing to minor cold straightening.

(iv) WING AND TAIL SURFACE RIBS. Damaged aluminum alloy ribs either of the stamped sheet-metal type or the built-up type employing special sections, square or round tubing, may be repaired by the additions of suitable reinforcement. Acceptable methods of repair are shown in Figures 3–17 and 3–18. These examples deal with types of ribs commonly found in small and medium aircraft. Any other method of reinforcement should be specifically approved by a representative of the Civil Aeronautics Administration.

(*a*) TRAILING AND LEADING EDGES AND TIP STRIPS. Repairs to wing and control surface trailing and leading edges and tip strips should be made by properly executed and reinforced splices. Acceptable methods of trailing edge repairs are shown in Figure 3–19.

(v) REPAIR OF DAMAGED SKIN.

(*a*) REPLACEMENT OF PORTIONS OF SKIN PANELS. In case metal skin is damaged extensively, 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, bulkheads, etc., and each seam should be made exactly the same in regard to rivet size, spacing, and rivet pattern as the parallel manufactured seams at the edges of the original sheet. If the two manufactured seams are different, the stronger one should be copied. See Figure 3–20 for typical acceptable methods of repairs.

(*b*) PATCHING OF SMALL HOLES. Small holes in skin panels which do not involve damage to the stiffening members may be patched by covering the hole with a patch plate in the manner shown in Figure 3–20.

(vi) SPLICING OF SHEETS. In some cases the method of copying the seams at the edges of a sheet may not be satisfactory; for example, when the sheet has cutouts, or doubler plates at an edge seam, or when other members transmit loads into the sheet. In these cases, the splice should be designed as illustrated in the following example:

Material: Clad 17S sheet, 0.032 inch thickness. Width of sheet (i. e. length at splice) = "W" = 10 inches.

To determine rivet size and pattern for a single-lap joint, similar to Figure 3–14:

(*a*) Use rivet diameter of approximately three times the sheet thickness. $3 \times 0.032 = 0.096$ inch. Use $\frac{1}{8}$ A17S–T3 rivets ($\frac{5}{32}$ A17S–T3 would also be satisfactory).

(*b*) Determine the number of rivets required per inch of width, "W", from Table 3–5. Number per inch $= 4.9 \times .75 = 3.7$. Total number of rivets required $= 10 \times 3.7 = 37$ rivets.

(*c*) Lay out rivet pattern with spacings not less than those shown in Figure 3–14. Referring to Figure 3–14A, it is seen that a double-row pattern with the minimum spacing will give a total of 40 rivets. However, as only 37 rivets are required, two rows of 19 rivets each, equally spaced over the 10 inches will result in a satisfactory splice.

(vii) STRAIGHTENING OF STRINGERS OR INTERMEDIATE FRAMES.

(*a*) MEMBERS SLIGHTLY BENT. Members which are slightly bent may be straightened cold and examined with a magnifying glass for injury to the material. The straightened parts should then be reinforced to an extent depending upon the condition of the material and the magnitude of any remaining kinks or buckles. If any strain cracks are apparent, complete reinforcements should be added by following the manufacturer's recommendations and the attachment of the reinforcements should be made in sound metal beyond the damaged portion.

(*b*) LOCAL HEATING. Local heating should never be applied to facilitate bending, swaging, flattening, or expanding operations on heat-treated aluminum alloy members, as it is difficult to control the temperature closely enough to prevent possible damage to the metal and it may impair its corrosion resistance. However, a torch with a large, soft flame is sometimes played over the surface of the cold worked aluminum of the nonheat-treatable alloys to anneal for bending or forming. This practice is permissible for these types of alloys when it is impracticable to anneal in a furnace or bath. The metal should not be heated above a temperature indicated by the charring of a resinous pine stick.

(*viii*) SPLICING OF STRINGERS AND FLANGES. Splices should be made in accordance with the manufacturer's recommendations, which are usually contained in a repair manual.

Typical splices for various shapes of sections are shown in Figures 3–21 and 3–23. Splices should be designed to carry both tension and compression and the splice shown in Figure 3–22 will be used as an example illustrating the following general principles:

(*a*) STATEMENT OF PRINCIPLES.

(*1*) To avoid eccentric loading and consequent buckling in compression, splicing or reinforcing parts should be placed as symmetrically as possible about the centerline of the member and attachment made to as many elements as necessary to prevent bending in any direction.

(*2*) To avoid reducing the strength in tension of the original bulb angle, the rivet holes at the ends of the splice are made small (no larger than the original skin attaching rivets), and the second row of holes (those through the bulbed leg) are staggered back from the ends. In general the rivets should be arranged in the splice so that the design tensile load for the member and splice plate can be carried into the splice without failing the member at the outermost rivet holes.

(*3*) To avoid concentration of load on the end rivet and consequent tendency toward progressive rivet failure, the splice is tapered off at the ends, in this case by tapering the backing angle and by making it shorter than the splice bar (see Figure 3–22).

The preceding principles are especially important in splicing stringers on the lower surface of stressed skin wings, where high tension stresses may exist. When several adjacent stringers are spliced, the splices should be staggered if possible.

(*b*) SIZE OF SPLICING MEMBERS. When the same material is used for the splicing member as for the original member, the net cross sectional area (i. e., the shaded areas in Figure 3–21) of the splicing member should be greater than the area of the section element which it splices. The area of a section element (e. g. each leg of an angle or channel) is equal to the width multiplied by the thickness. For example, in Figure 3–22, the bar, "B" is assumed to splice the upper leg of the stringer, and the angle "A" to splice the bulbed leg of the stringer. Since the splice bar "B" is not as wide as the adjacent leg, and since the rivet diameter is also subtracted from the width, the bar is made twice as thick in order to obtain sufficient net area.

(*c*) THE DIAMETER OF RIVETS IN STRINGERS. The diameter of rivets in stringers should preferably be between two and three times the thickness, "t", of the leg, but should not be more than one-fourth the width, "W", of the leg. Thus, 1/8-inch rivets are chosen in the example, Figure 3–22. If this splice were in the lower surface of a wing, the end rivets would be made the same size as the skin attaching rivets, say 3/32.

(*d*) THE NUMBER OF RIVETS. The number of rivets required on each side of the cut in a stringer or flange may be determined from standard textbooks on aircraft structures, or may be found from Tables 3–4, 3–5 or 3–6 as the case may be. In determining the number of rivets required in the example, Figure 3–22, for attaching the splice bar, "B", to the upper leg, the thickness "t" of the element of area being spliced is 1/16 inch (use 0.064), the rivet size is 1/8 inch, and Table 3–5 shows that 9.9

rivets are required per inch of width. Since the width, "W", is ½ inch, the actual number of rivets required to attach the splice bar to the upper leg, *on each side of the cut*, is 9.9 (rivets per inch) × .5 (inch width) = 4.95; use 5 rivets.

For the bulbed leg of the stringer, "t" = $\frac{1}{16}$ inch (use 0.064), AN–3 bolts are chosen, and the number of bolts required per inch of width = 3.3. The width, "W", for this leg, however, is 1 inch, and the actual number of bolts required on each side of the cut is 1 × 3.3 = 3.3; use 4 bolts. When both rivets and bolts are used in the same splice, the bolt holes should be accurately reamed to size. It is preferable to use only one type of attachment, but in the above example, the dimensions of the legs of the bulb angle indicated rivets for the upper leg and bolts for the bulb leg.

(*e*) SPLICING OF INTERMEDIATE FRAMES. The same principles that are used for stringer splicing may be applied to intermediate frames, when the following point is also considered:

Conventional frames of channel or Z section are relatively deep and thin compared to stringers, and usually fail by twisting or by buckling of the free flange. The splice joint should be reinforced against this type of failure by using a splice plate heavier than the frame and by splicing the free flange of the frame with a flange of the splice plate, as illustrated in Figure 3–24. Since a frame is likely to be subjected to bending loads, the length of splice plate "L" should be more than twice the width, "W_2", and the rivets spread out to cover the plate.

(ix) REPAIRING CRACKED MEMBERS. Acceptable methods of repairing various types of cracks occurring in service in structural elements from various causes are shown in Figures 3–25 to 3–28. The following general procedure should be followed in repairing such defects:

(*a*) Small holes $\frac{3}{32}$ inch (or ⅛ inch) should be drilled at the extreme ends of the cracks to prevent their spreading further.

(*b*) Reinforcements as shown in these figures should be added to carry the stresses across the damaged portion and to stiffen the joints.

The condition causing such cracks to develop at a particular point is stress concentration at that point in conjunction with repetition of stress (such as produced by vibration of the structure). The stress concentration may be due to the design or to defects such as nicks, scratches, tool marks, and initial stresses or cracks from forming or heat-treating operations. It should be noted that an increase in sheet thickness alone is usually beneficial but does not necessarily remedy the conditions leading to cracking.

(6) FITTINGS.

(i) STEEL FITTINGS.

(*a*) INSPECTION FOR DEFECTS. Fittings should be free from scratches, vise, and nibbler marks, and sharp corners. A careful examination of the fitting with a medium power (at least 10 power) magnifying glass will be considered an acceptable inspection.

When repairing aircraft after an accident or in the course of a major overhaul, all highly stressed main fittings should be inspected in accordance with the provisions of CAM 18.20–6, and, if necessary, corrosion prevention measures taken as recommended in CAM 18.20–7.

(*b*) TORN, KINKED, OR CRACKED FITTINGS. Torn, kinked, or cracked fittings should be replaced and not repaired.

(*c*) ELONGATED OR WORN BOLT HOLES. Elongated holes in fittings which were designed without bushings should not be reamed oversize but such fittings should be replaced unless the method of repair is approved by a representative of the Civil Aeronautics Administration. Holes should not be filled with welding rod. Acceptable methods of repairing elongated or worn bolt holes in landing gear, stabilizer, interplane or cabane strut ends only, not originally equipped with pin plates, are shown in Figure 3–29. (See also Figure 3–8 on longeron repair at a fitting.)

(ii) ALUMINUM AND ALUMINUM ALLOY FITTINGS. Damaged fittings should be replaced with new parts having the same material specifications or the method of repair should be specifically approved by a representative of the Civil Aeronautics Administration.

(7) CASTINGS. *Damaged castings* should be replaced and not repaired unless the method of repair is specifically approved by a representative of the Civil Aeronautics *Administration.*

18.20–4 CONTROL CABLES AND TERMINALS. ***(CAA policies which apply to section 18.20.)***

(a) CONTROL CABLES AND WIRES. Control cables and wires should be replaced if injured, distorted, worn or corroded even though the strands are not broken. However, cable sections may be spliced using the procedures of CAM 18.20–4 (a) (1).

(1) SPLICING. Control cables may be spliced when they become worn, distorted, corroded or otherwise injured. The cable, thimbles, shackles, turnbuckles, bolts, and other parts should be of the same size, material and quality as the original parts or of such size that the repaired cable will be of strength equivalent to the original. However, AN–666 through AN–669 standard swaged cable terminals develop the full cable strength and may be substituted for the original terminals wherever practical. If facilities and supplies are limited, replacement sometimes may be prepared, using thimbles, bushings, and turnbuckles in place of original terminals. When this is done, flexible cables 7 × 7 and 7 × 19, having a diameter of $\frac{3}{32}$ inch or over may be woven spliced by means of the five-tuck method. Flexible cable less than $\frac{3}{32}$ inch in diameter and nonflexible carbon steel 19 wire cable (AN–C–76) may be wrap-soldered. Directions for fabricating these splices and limitations as to their use are contained in the *following paragraphs.*

All splices should be installed so that no portion of the splice comes closer than two inches to any fairlead or pulley and the connections should not be located at points where jamming may occur during any portion of the travel of either the loaded cable or the slack cable in the deflected position.

(2) SUBSTITUTION OF CABLE. Substitution of control cable (aircraft cord) for hard or streamlined wires will not be acceptable unless specifically approved by a representative of the Civil Aeronautics Administration.

(3) CUTTING AND HEATING. Cables should be cut to length only by mechanical means. The use of a torch in any manner is not permitted. Wires and cables should never be subjected to excessive temperature. Soldering bonding braid to control cable will not be considered satisfactory.

(4) RUST PREVENTION. If the cables are made from tinned steel, a rust-preventative oil should be utilized to coat the cable. It is to be noted that corrosion-resistant steel cable does not require this treatment *for rust prevention.*

(b) SWAGED TERMINALS.

(1) AN–666 THROUGH AN–669. Standard swaged cable terminals, AN–666 through AN–669, may be used for replacement purposes. Any cable swaging tool is satisfactory for swaging AN terminals provided full cable strength is developed and "after swaging" dimensions meet those shown on the pertinent drawing.

(2) BALL - AND - SOCKET - TYPE TERMINALS. Ball-and-socket-type swaged terminals and other types that do not positively prevent cable unwinding should not be used for general replacement except where they were utilized on the original installation by the aircraft manufacturer.

(c) WOVEN SPLICE TERMINAL. The five-tuck woven splice terminals shown *in Figure 4–1 may be utilized on* 7 × 7 flexible and 7 × 19 extra flexible cables of $\frac{3}{32}$ inch diameter or greater. This type of terminal will develop only 75 percent of the cable strength and should not be used to replace swaged or other high efficiency terminals unless it is definitely known that the design load for the cable is not greater than 75 percent of the cable minimum breaking strength (see Table 4–1).

In some cases it will be necessary to splice one end of the cable on assembly. For this reason, investigate the original installation for pulleys and fairleads that might restrict the passage of the splice. The procedure for the fabrication of a woven splice is as follows: See Figure 4-1 for the designation of numbers and letters referred to in this sequence of operations.

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

(2) Select the free strand (1) nearest the standing length at the end of the fitting and free this strand from the rest of the free ends. Next, insert a marlinspike under the first three strands (A, B, C) of the standing length nearest the separated strand of the free end and separate them momentarily by twisting the marlinespike. Insert the free strand (1) under the three separated strands through the opening created by the marlinespike. Pull the free end taut by means of pliers.

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

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

(5) Begin with the first free strand (1) and work in a counterclockwise direction, tucking free strands under every other standing strand. After the completion of every tuck, pull the strands tight with pliers. Pull toward the end fitting (see detail C). After the completion of the third complete tuck, cut in half the number of wires in each free strand. Make another complete tuck with the wires remaining. At the completion of the fourth tuck, again halve the number of wires in the free strands and make one final tuck with the wires remaining. Cut off all protruding strands and pound the splice with a wooden or rawhide mallet to relieve the strains in the wires. Serve the splice with waxed linen cord (6 ply, type B, Federal Specification V-T-291).

Start 1/4 inch from the end of the splice and carry the wrapping over the loose end of the cord and along the tapered splice to a point between the second and third tucks. Insert the end of the cord back through the last five wrappings and pull snug. Cut off the end, and if a thimble is used as an end fitting bend down the points. Apply two coats of shellac to the cord, allowing 2 hours between coats (see detail D). Carefully inspect the cable strands and splices for local failure. Weakness in a woven splice is made evident by a separation of the strand of serving cord.

(d) WRAP-SOLDERED SPLICE. The wrap-soldered splice terminal shown in Figure 4-2 may be utilized on flexible cables less than 3/32 inch in diameter and on nonflexible single-strand (19-wire) cable (AN-C-76). This type of terminal will develop only 90 percent of the cable strength and should not be used to replace swaged or other high efficiency terminals unless it is definitely known that the design load for the cable is not greater than 90 percent of the cable minimum breaking strength (see Table 4-1).

The method of making the wrapped and soldered splice is as follows:

(1) The serving or wrapping wire shall be of commercial soft-annealed steel wire or

commercial soft iron wire, thoroughly and smoothly tinned or galvanized.

(2) The solder shall be half-and-half tin and lead conforming to Federal Specification QQ-S-571. The melting point of this solder varies from 320° to 390° F., and the tensile strength is approximately 5,700 pounds per square inch.

(3) Solder flux shall be a compound of stearic acid (there shall be no mineral acid present) and resin, with a composition of 25 to 50 percent stearic acid and 75 to 50 percent resin. A warming gluepot to keep the flux in fluid state is desirable.

(4) Before the cable is cut the wires are soldered to prevent slipping. The preferable process is to tin and solder the cable thoroughly for 2 or 3 inches by placing in a solder trough, finishing smooth with a soldering tool. The cable may be cut diagonally to conform to the required taper finish.

(5) After being soldered and cut the cable is securely bent around the proper size thimble and clamped, taking care that the cables lie close and flat and that the taper end for finish lies on the outside. If it is necessary to trim the taper at this point in the process, it is preferable that it be done by nipping, but grinding is permissible, provided a steel guard at least 3 inches long and 1/32 inch thick is placed between the taper end and the main cable during the operation, and that the heat generated from the grinding does not melt the solder and loosen the wires.

(6) Serving may be done by hand or machine, but in either case each serving convolution must touch the adjoining one and be pulled tightly against the cable, with spaces for permitting a free flow of solder, and inspection.

(7) Care must be exercised to prevent drawing of the temper of any cable wires by excessive temperature or duration of applied heat. The flux used in this soldering shall be stearic acid and resin. The use of salammoniac or other compounds having a corrosive effect is not permitted either as a flux or for cleaning the soldering tools.

(8) Soldering is accomplished by immersing the terminal alternately in the flux and in the solder bath, repeating the operation until thorough tinning and filling with solder under the serving wire and thimble is obtained. The temperature of the solder bath and place where terminal is withdrawn shall not be above 450° F. A soldering iron may be used in the final operation to give a secure and good-appearing terminal. Care must be taken that the solder completely fills the space under the serving wire and thimble. A slightly hollowed cast iron block to support the splice during soldering may help in securing the best results. The use of abrasive wheels or files for removing excess solder is not permissible.

(9) As an alternative process for making terminals for nonflexible cable, the oxyacetylene cutting method and the presoldering method (soldering before wrapping) are permitted, but only on the following conditions: (i) that the process of cutting securely welds all wires together; (ii) that the annealing of the cable does not extend more than one cable diameter from the end; (iii) that no filing be permitted either before or after soldering; (iv) for protection during the operation of grinding the tapered end of the cable, a steel guard at least 3 inches in length and 1/32 inch thick shall be placed between the taper and the main cable; (v) the heat from grinding shall not draw the temper of the cable.

(10) Wrap-soldered splice terminals shall not be used ahead of the firewall, or in other fire zones, or in other locations where they might be subjected to high temperature.

(e) SAFETYING OF TURNBUCKLES. All turnbuckles should be safetied with safety wire using either the double or single wrap method. For safety wire sizes and materials refer to Table 4-2. *Safety wire should never be reused.* The turnbuckle should be adjusted to the correct cable tension so that not more than three threads are exposed on either side of the turnbuckle barrel. Turnbuckles should never be lubricated.

(1) DOUBLE WRAP METHOD.

(i) This method is preferred as offering the best method of safetying turnbuckles. Two separate lengths of the proper wire should be used. One length of wire is to be run through the hole in the barrel of the turnbuckle and the ends of the wire bent towards opposite ends of the turnbuckle. The second length of wire is then to be passed through the hole in the turnbuckle barrel and the ends bent along the barrel on the opposite side from the first length. Spiral the first two wires in opposite directions around the barrel so as to cross each other twice. The wires at one end of the turnbuckle are passed through the eye in opposite directions and one wire laid along the barrel, while the other wire is wrapped at least four turns around the shank of the turnbuckle and the wire alongside the barrel before cutting off the end. The remaining length of safety wire should then be wrapped at least four turns around the shank of the turnbuckle and cut off. Repeat this same wrapping procedure with the wire ends at the opposite end of the turnbuckle. This method of double wrap safetying is shown in Figure 4-3 (A).

(ii) Another double wrap method which is satisfactory is done as in (i) except that the spiraling of the wires is omitted as shown in Figure 4-3 (B).

(iii) The double wrap procedure given in Navy Specification PO-42A, Amendment No. 1, may be used as a replacement only when utilized as an original installation by the aircraft manufacturer.

(2) SINGLE WRAP METHOD. The single wrap method is acceptable but is not the equal of the double wrap method.

(i) Pass a single length of wire through the cable eye or fork at either end of the turnbuckle assembly and spiral each of the wire ends in opposite directions around the first half of the turnbuckle barrel so as to cross each other three times. Both of the wire ends are threaded together through the hole in the middle of the turnbuckle so that the third crossing of the wires is in the center of the hole. The two wire ends are then again spiraled in opposite directions around the remaining half of the turnbuckle crossing each other twice. One of the wires is then passed through the cable eye or fork and both wires are then wrapped around the shank for at least four turns each. This method is shown in Figure 4-3 (C).

(ii) An acceptable single wrap may be made in the following manner. One length of wire is passed through the center hole of the turnbuckle and the ends bent toward opposite ends of the turnbuckle. The wire is then passed through the turnbuckle eye and wrapped around the shank as shown in Figure 4-3 (D). After safetying the turnbuckle, no more than three threads should be exposed, and the ends of each safety wire should be securely fastened by at least four wraps.

18.20-5 BOLTS, SCREWS, AND MISCELLANEOUS FASTENERS. ***(CAA policies which apply to section 18.20.)***

(a) BOLTS. Most bolts used in aircraft structures are either general-purpose AN bolts, or NAS (National Aircraft Standard) internal-wrenching or close-tolerance bolts. In certain cases aircraft manufacturers make up special bolts for a particular application and it is important to use like or better bolts in replacement.

(1) IDENTIFICATION. AN-type aircraft bolts can be identified by the code markings on the bolt heads. The markings generally denote the bolt manufacturer, the material of which the bolt is made, and whether the bolt is a standard AN-type or a special purpose bolt.

AN standard steel bolts are marked with either a raised dash or asterisk, corrosion-resistant steel is indicated by a single raised dash, and AN aluminum alloy bolts are marked with two raised dashes. The strength and dimensional details of AN bolts are specified on the Army-Navy Aeronautical Standard Drawings.

Special purpose bolts include the high-strength and low-strength types, close-tolerance types, and bolts inspected by magnetic or fluorescent means.

Typical markings include "SPEC" (usually highly heat treated), an aircraft manufacturer's part number stamped on the head, or plain heads (low strength). Close-tolerance National Aircraft Standards (NAS) bolts are marked with either a raised or recessed triangle. The material markings for NAS bolts are the same as for AN bolts, except that they may be either raised or recessed. Bolts inspected magnetically (magnaflux) or by fluorescent means (Zyglo) are identified by means of colored lacquer or a head marking of a distinctive type. The aircraft manufacturer or the CAA should be contacted when information is needed on a particular special bolt. Figure 5-1 shows the typical coding used on aircraft bolt heads.

(2) GRIP LENGTH. In general, bolt grip lengths should be equal to the material thickness. However, bolts of slightly greater grip length may be used provided washers are placed under the nut or the bolt head, or in the case of plate nuts, shims may be added under the plate. For proper washers refer to CAM 18.20-5 (e).

(3) LOCKING OR SAFETYING OF BOLTS. All bolts and/or nuts, except self-locking nuts, should be suitably locked or safetied. Cotter-pins and safety wire should not be used a second time.

(4) BOLT FIT. All bolt holes should have close tolerances. Generally, it is permissible to use the first lettered drill size larger than the nominal bolt diameter except where the AN hexagon bolts are used in light-drive fit (reamed) applications and where NAS close-tolerance bolts or AN clevis bolts are used. Bolt holes should be normal to the surface involved to provide full bearing surface for the bolt head and nut and should not be oversize or elongated. In case of oversize or elongated holes a qualified agent or engineer should always be consulted before drilling or reaming the hole to take the next larger bolt, as usually items such as edge distance, clearance, etc., must be considered.

(5) TORQUES. Table 5-1 gives the recommended torque values for both the fine and coarse thread series of nuts. These values should be followed unless other instructions require a specific torque for a given nut. Bolts and nuts should be clean, dry, and thoroughly degreased before installation. Threads should not be oiled as this will destroy the torque values shown in the table. Nuts should be torqued in all important places, such as wing-joint bolts, engine-support bolts, landing-gear bolts, and the like.

When tightening castellated nuts on bolts, it is possible that the cotter-pin holes will not line up with the slots in the nuts for the range of recommended values listed in Table 5-1. In such cases the nut may be over-tightened just enough to line up the next slot with the cotter-pin hole even though the maximum tabulated torque value is exceeded.

(6) HEX-HEAD BOLTS (AN-3 through AN-20). The hex-head aircraft bolt is an all-purpose structural bolt used for general applications involving tension or shear loads. Alloy steel bolts smaller than No. 10-32 and aluminum alloy bolts smaller than ¼-inch diameter should not be used in primary structure. Aluminum alloy bolts and nuts should not be used where they will be repeatedly removed for purposes of maintenance and inspection. Aluminum alloy nuts may be used on cadmium-plated steel bolts in shear in land airplanes, but should not be used on seaplanes due to electrolytic action between dissimilar metals.

(7) CLOSE-TOLERANCE BOLTS (NAS-53 through NAS-66) (HEX-HEAD), AND (NAS-80 through NAS-86) (100° COUNTERSUNK). Close-tolerance bolts are used in high-performance aircraft in applications where the bolted joint is subject to severe load reversals and vibration. The standard AN hex-head bolts may be used for the same applications provided a light-drive fit is accomplished.

(8) INTERNAL - WRENCHING BOLTS (NAS-144 through NAS-158). The internal-wrenching bolt is a high-strength bolt suitable for use both in tension and shear applications. In steel parts, either the bolt hole should be countersunk to seat the large radius of the shank at the head or, as in dural, a special heat-treated washer (NAS-143C) should be used to fit the head and to provide adequate bearing area. A special heat-treated plain washer (NAS-143) is used under the nut. Special nuts

should be used on these bolts. (Refer to CAM 18.20-5 (d) (3) (vii)).

Standard AN hex-head bolts and washers cannot be substituted for them, as they do not have the required strength.

(9) DRILLED-HEAD BOLTS (AN-73 through AN-81). The AN drilled-head bolt is similar to the standard hex-bolt, but has a deeper head which is drilled to receive wire for safetying. The AN-3 and the AN-73 series of bolts are interchangeable for all practical purposes from the standpoint of tension and shear strengths.

(b) SCREWS. In general, screws differ from bolts by the following characteristics: usually lower material strength, a looser thread fit (number 2), head shapes formed to engage a screw driver, and the shank threaded along its entire length without a clearly defined grip. However, several types of structural screws are available that differ from the standard structural bolts only in the type of head. The material is equivalent and a definite grip is provided. The AN-525 washerhead screws, the AN-509 structural screws, and the NAS-204 through NAS-235 are such parts. The material markings are the same as those used on AN standard bolts.

(1) STRUCTURAL SCREWS (NAS-204 through NAS-235) (AN-509 AND AN-525). This type of screw, when made of alloy steel, such as 4130, NE-8630, or equivalent, and heat-treated from 125,000 to 145,000 p.s.i., may be used for structural assembly in shear applications similar to structural bolts.

(2) SELF-TAPPING SCREWS. The AN-504 and AN-506 screws are used for attaching minor removable parts such as nameplates and the like. AN-530 and AN-531 are used in blind applications for the temporary attachment of sheet metal for riveting and the permanent assembly of nonstructural assemblies. AN-535 is a plain head self-tapping screw used in the attachment of nameplates or in sealing drain holes in corrosion-proofing tubular structures and is not intended to be removed after installation. Self-tapping screws should never be used to replace standard screws, nuts, bolts, or rivets in the original structure.

(c) PINS. The three types of pins used in aircraft structures are: the taper pin, the flathead pin, and the cotter pin. Pins are used in shear applications and for safetying.

(1) Taper pins (AAF-385 and AN-386) plain and threaded, are used in joints which carry shear loads and where absence of play is essential. The plain taper pin is drilled and usually safetied with wire. The threaded taper pin is used with a taper-pin washer (AN-975) and shear nut (safetied with cotter pin) or self-locking nut.

(2) The flathead pin (AN-392 through AN-406) commonly called a clevis pin, is usually used in conjunction with tie rod terminals and in secondary controls which are not subjected to continuous operation. The pin should be safetied with a cotter pin and is customarily installed with the head up so that if the cotter pin fails or works out, the pin will remain in place.

(3) The AN-380 cotter pin is used for safetying bolts, screws, nuts, other pins, and in various applications where such safetying is necessary. The AN-381 cotter pin is used in locations where nonmagnetic material is required, or in locations where resistance to corrosion is desired.

(d) NUTS.

(1) SELF-LOCKING NUTS. Self-locking nuts are acceptable for use on certificated aircraft subject to the restrictions on the pertinent "Manufacturers' Recommended Practice Sheets." Self-locking nuts are used on aircraft to provide tight connections which will not shake loose under severe vibration. Two types of self-locking nuts are currently in use, the all-metal type and the fibre-lock type. Self-locking nuts should not be used at joints which subject either the nut or bolt to rotation. They may be used with antifriction bearings and control pulleys provided the inner race of the bearing is clamped to the supporting structures by the nut and bolt. Nuts which are attached to the structure should be attached in a posi-

tive manner to eliminate rotation or misalinement when tightening the bolts or screws.

(i) All-metal nuts are constructed with either the threads in the locking insert out-of-phase with the load-carrying section or with a saw-cut insert with a pinched-in thread in the locking section. The locking action of the all-metal nut depends upon the resiliency of the metal when the locking section and load-carrying section are engaged by screw threads.

(ii) Fiber-lock nuts are constructed with an unthreaded fiber locking insert held securely in place. The fiber has a smaller diameter than the nut, and when a bolt or screw is entered, it taps into the fiber producing a locking action. After the nut has been tightened one full thread of the bolt or screw should extend through the nut. When fiber-type self-locking nuts are reused, care should be exercised that the fiber has not lost its locking friction or become brittle. Fiber-type self-locking nuts should not be reused if they can be run up finger tight. Bolts $5/16$-inch diameter and over with cotter-pin holes may be used with self-locking nuts but only if free from burrs around the holes. Bolts with damaged threads and rough ends should never be used. Do not tap the fiber-locking insert.

(iii) Self-locking nut bases are made in a number of forms and materials for riveting and welding to aircraft structure or parts.

Certain applications require the installation of self-locking nuts in channels, an arrangement which permits the attachment of many nuts with only a few rivets. These channels are track-like bases with regularly spaced nuts which are either removable or nonremovable. The removable type carries a floating nut which can be snapped in or out of the channel thus making possible the ready removal of damaged nuts. Nuts such as the clinch-type and spline-type which depend on friction for their anchorage are not acceptable for use in aircraft structures.

(iv) Self-locking nuts may be used on aircraft engines and accessories when their use is specified by the engine manufacturer in his bulletins or manuals. Refer to CAM 18.20-13 for detailed installation instructions.

(2) AIRCRAFT CASTLE NUT (AN-310). The castle nut is used with drilled shank AN hex-head bolts, clevis bolts, eye bolts, drilled-head bolts or studs, and is designed to accommodate a cotter pin or lock wire as a means of safetying.

(3) MISCELLANEOUS AIRCRAFT NUTS.

(i) The plain nut (AN-315 and AN-335) has limited use on aircraft structures and requires an auxiliary locking device such as a check nut or lock washer.

(ii) Light hex nuts (AN-340 and AN-345) are used in miscellaneous applications and must be locked by an auxiliary device.

(iii) The check nut AN-316 is used as a locking device for plain nuts, screws, threaded rod ends and other devices.

(iv) The castellated shear nut AN-320 is designed for use with clevis bolts and threaded taper pins, which are normally subjected to shearing stress only.

(v) Wing nuts AN-350 are intended for use on hose clamps and battery connections, etc., where the desired tightness is ordinarily obtained by the use of the fingers or hand tools.

(vi) Sheet spring nuts, such as speed nuts, are used with standard and sheet metal self-tapping screws in nonstructural locations. They find various uses in supporting line clamps, conduit clamps, electrical equipment, access doors, and the like, and are available in several types.

(vii) Two commercial types of internal wrenching nuts are available, the internal wrenching Elastic Stop Nut and the Unbrako internal-wrenching nut. Both are of the self-locking type, are heat-treated, and are capable of carrying the high strength bolt tension load.

(e) WASHERS. The types of washers used in aircraft structure are: plain washers, lock washers, and special washers.

(1) Plain washers AN-960 and AN-970 are widely used under hex nuts to pro-

vide a smooth bearing surface, to act as a shim and to adjust the position of castellated nuts with respect to drilled cotter-pin holes in bolts. Plain washers should be used under lock washers to prevent damage to surfaces. Cadmium-plated steel washers should be used under bolt heads or nuts on aluminum alloy or magnesium structures where corrosion will then be between the washer and the steel. The AN-970 steel washer provides a greater bearing area than the plain type and is used in wooden structures under both bolt heads and nuts to prevent local crushing of the surface.

(2) Lock washers AN-935 and AN-936 may be used with machine screws or bolts whenever the self-locking or castellated type of nut is not applicable. They should not be used as fastenings to primary or secondary structures or where subject to frequent removal or corrosive conditions.

(3) Ball-socket and seat-washers AC-950 and AC-955 are used in special applications where the bolt is installed at an angle to the surface, or where perfect alinement with the surface is required at all times. These washers are used together.

(4) Taper-pin washers AN-975 are used with the threaded taper pin.

(5) NAS-143 washers for internal-wrenching nuts and bolts are used with NAS internal-wrenching bolts. Type "C" is countersunk to seat the bolthead shank radius and a plain-type washer is used under the nut. Both of these washers are heat-treated from 125,000 to 145,000 p.s.i.

(f) RIVETS.

(1) STANDARD SOLID-SHANK RIVETS. The universal-head rivets AN-470 are used in aircraft construction in both interior and exterior locations.

Round-head rivets AN-430 and AN-435 are used in the interior of aircraft except where clearance is required for adjacent members.

Flat-head rivets AN-441 and AN-442 are used in the interior of the aircraft where interference of adjacent members does not permit the use of round-head rivets.

Brazier-head rivets AN-455 and AN-456 are used on the exterior surfaces of aircraft where flush riveting is not essential.

Countersunk-head rivets AN-426 (100 degrees), AN-425 (78 degrees), and AN-420 (90 degrees) are used on the exterior surfaces of aircraft to provide a smooth aerodynamic surface, and in other applications where a smooth finish is desired.

(i) MATERIAL APPLICATIONS.

(*a*) The A-17S-T3 rivet is the most commonly used rivet material utilized in aluminum alloy structures. Its main advantage lies in the fact that it may be used in the condition received without any further treatment.

(*b*) The 17S-T3 and 24S-T4 rivets are used in aluminum alloy structures where strength higher than that of the A17S-T3 rivet is needed.

(*c*) The 2S rivet of pure aluminum is used for riveting nonstructural parts fabricated from the softer aluminum alloys, such as 2S, 3S, and 52S.

(*d*) The 56S rivet is used for riveting magnesium alloy structures because of its corrosion-resistant qualities in combination with magnesium. No other rivets should be used for this purpose.

(*e*) Mild steel rivets are used primarily in riveting steel parts. Galvanized rivets should not be used on steel parts subjected to high heat.

(*f*) Corrosion-resistant steel rivets are used primarily in riveting corrosion-resistant steel parts, such as firewalls, exhaust stack bracket attachments and similar structures.

(*g*) Monel (nickel steel alloy) rivets are used in special cases for riveting high nickel steel alloys and nickel alloys. Monel rivets may be used interchangeably with stainless steel rivets, and are more easily driven. However, it is preferable to use stainless steel rivets in stainless steel parts.

(*h*) Copper rivets are used for riveting copper alloys, leather, and other nonmetallic materials. This rivet has only limited usage in aircraft.

(2) BLIND RIVETS. Blind rivets may be substituted for the normally required

solid rivets in accordance with the blind rivet manufacturer's recommendations. They should not be used where the failure of a few rivets will seriously impair the airworthiness of the aircraft.

(g) FASTENERS (COWL AND FAIRING). A number of patented fasteners are in use on aircraft. Among them are Dzus, Shakeproof, Quick Lock, Airlock, Camloc, and others. The manufacturer's recommendations concerning the proper use of these types of fasteners should always be considered in other than replacement applications.

(h) UNCONVENTIONAL ATTACHMENTS. "Rivnuts," "Lok-skrus," and other unconventional or new attachment devices should not be used in the primary structure unless approved by a representative of the Civil Aeronautics Administration.

18.20–6 CORROSION TREATMENT, CLEANERS, AND PAINT REMOVERS. *(CAA policies which apply to section 18.20.)*

(a) CORROSION PROTECTION.

(1) GENERAL CORROSION. Almost all metals used in aircraft are subject to corrosion. Materials such as steel will rust, and aluminum and magnesium will form corrosion products, unless properly protected. Stainless steel, brasses, and copper alloys normally form a surface film which prevents further surface corrosion; however, under certain conditions particularly when in contact with dissimilar metals, even these alloys must be protected. General corrosion is always promoted by contact of metals with materials that absorb water. For example: wood, sponge rubber, felt, etc., may be sources of serious corrosion unless proper protection is used.

(2) DISSIMILAR METALS CORROSION. When two dissimilar metals are in contact and are connected by an electrolyte (water), corrosion may occur. For this reason metals have been divided into certain groups, based on their susceptibility to this form of corrosion. Unprotected contact between metals of different groups may result in dissimilar metals corrosion; therefore, contact between metals of dissimilar groups should be prevented or the contact surface should be adequately protected.

Similar metal groups (refers to *surface* of metal).

Group 1	Magnesium alloys.
Group 2	Zinc, cadmium, lead, tin, steel.
Group 3	Copper and its alloys, nickel and its alloys, chromium, and stainless steels.
Group 4	All aluminum alloys.

Aluminum alloys (Group 4) may be further subdivided into the following subgroups:

Subgroup A.	2S, 3S, 52S, 53S, 61S, 75S, 13, 43, 220, 355, 356, all clad alloys such as Alclad, and Pureclad.
Subgroup B.	195, 14S, 17S, 24S.

Under severe corrosive conditions, the above subgroups should be considered as dissimilar metal groups insofar as corrosion protection is concerned. This is particularly true where a relatively large area of an alloy classified in subgroup B is in contact with a relatively small area of Subgroup A, in which case severe corrosion of the Subgroup A alloy may be expected.

(3) INTERGRANULAR CORROSION. Intergranular corrosion occurs in certain aluminum alloys which are improperly heat-treated. For example, 24S alloys must be quenched quickly after heat-treatment in order to prevent intergranular corrosion. Since 24S alloy contains metals other than aluminum, particularly copper, severe corrosion may result if this alloy is quenched slowly, and a reduction in strength may result in improperly quenched 24S alloys when subjected to corrosive conditions. This type of corrosion is difficult to detect in its original stage except by microscopic examinations. When well advanced it is characterized by scaling and blistering. Surface protection of slowly quenched 24S alloy will result in prevention of intergranular corro-

sion. The only protection which is adequate for air quenched 24S is cladding of the aluminum alloy with pure aluminum (such as Alcald and Pureclad sheet). It should be noticed that in some cases even clad alloys may be susceptible to intergranular corrosion. Other surface protection such as anodizing and subsequent coatings such as zinc chromate primer, heavy greases, etc., may also prevent intergranular corrosion in cases where susceptibility of the alloy to intergranular corrosion is not too great.

(4) STRESS CORROSION. This type of corrosion occurs when certain metals, mostly aluminum and magnesium alloys, are exposed to high stress and corrosive conditions. Stress corrosion has occurred in aluminum when steel bushings were pressed into the aluminum parts with too tight a fit, and were exposed to corrosive conditions. Stress corrosion can also occur in cold worked metals which are not properly stress relieved.

(b) CORROSION PROTECTION MEASURES. Surfaces which are completely dry cannot corrode. If a metal can be protected from moisture due to rain, condensation, or other causes, corrosion need not be feared.

Dirt, surface film, etc., on metal surfaces tend to retain moisture and hence promote corrosion. Water absorbing materials such as certain cleaners and calcium chloride which may occasionally be used as a snow remover on runways are especially dangerous in this regard.

(1) ANODIZING AND RELATED PROCESSES. In anodizing, aluminum alloys are placed in an electrolytic bath causing a thin film of aluminum oxide to form on the surface of the aluminum. This film is resistant to corrosion and affords a good paint base. Other processes which do not provide as good a corrosive protection as anodizing are, however, good paint bases. These processes are:

Alakaline cleaning followed by chromic acid dip.
Alcoholic phosphoric acid cleaner.
Alkaline dichromate treatment.

(2) PLATING. Steels are commonly plated with other metals to prevent corrosion. Plating is accomplished by placing the article in an electrolytic bath and metal from the plating solution is deposited on it. Various metals used in plating vary in the corrosion protection they afford steel. For instance, cadmium and zinc corrode before the steel does, hence slight breaks or cracks through plating of these metals will not result in rusting of the exposed steel, since the plated metal is corroded and protects the steel. Chromium does not protect steel by this method, as steel will corrode before the chromium does, and thus depends for its protection on the tightness of the plate.

(3) PARKERIZING AND BONDERIZING. These processes do not appear to be equal in corrosion protection to plating and are not generally acceptable as a substitute for plating; however, both are good paint bases.

(4) MAGNESIUM ALLOY TREATMENTS

(i) DICHROMATE TREATMENT FOR MAGNESIUM. The dichromate treatment consists of boiling magnesium parts in a solution of sodium dichromate resulting in a coating with little resistance to corrosion but which is a good paint base.

(ii) CHROMIUM PICKLE TREATMENT FOR MAGNESIUM. In this process the magnesium parts are placed in a solution of nitric acid and sodium dichromate. This will protect the magnesium during storage and acts as a bond for subsequent organic finishes.

(iii) GALVANIC ANODIZING TREATMENT FOR MAGNESIUM. This is an electrolytic process used to provide a paint base and corrosion preventive film on magnesium alloys containing manganese.

(5) CLADDING. Aluminum alloys which are susceptible to corrosion are frequently clad with pure aluminum. Slight pits, scratches, or other defects through the cladding material will not result in corrosion of the core, since the pure aluminum on the edges of the defect will be preferentially corroded, protecting the core.

(6) METAL SPRAYING. In this process metallic wire such as aluminum or zinc is fed into a special spray gun. The metal is melted and sprayed on the object to be protected, which must be thoroughly clean to prevent peeling of the sprayed coat. A "metallized" surface has very good resistance to corrosion if properly applied and of sufficient thickness.

(7) ORGANIC COATINGS. Zinc chromate primer, enamels, chlorinated rubber compounds, etc., are organic coatings commonly used on metals to protect them. The finishes should be applied according to the instructions of the manufacturer.

(8) DOPE-PROOFING. When doped fabrics are applied over an organic finished metal structure, the dope will have a tendency to loosen the finish on the metal. For this reason, organic coatings on the metal are usually covered with a dope-proof paint, with metal foil, or with cellulose tape to prevent the dope from striking through.

(9) TUBE INTERIORS. The interiors of structural steel and aluminum tubing should be protected against corrosion. A small amount of water entrapped in a tube can corrode entirely through the tube thickness in a short period of time. For this reason, most structural tubing is flushed with hot linseed oil, paralketone, or other corrosion inhibitor. Hot flushing results in a good coating. The flushing liquid is usually introduced through small holes drilled in the tubing. These holes should be plugged with a PK screw or by other means to prevent entry of moisture. Air and watertight sealing of the tubing will also give adequate protection against corrosion if the tubing is internally dry before being sealed.

(c) CORROSION-PROOFING OF LANDPLANES, SEAPLANES, AND LANDPLANES CONVERTED TO SEAPLANES.

(1) LANDPLANES AND SEAPLANES. In the repair or alteration of aircraft, corrosion-proofing the same or equivalent to that originally applied should be used unless the repair or alteration would cause an increase in liability to corrosion, in which case additional corrosion protection measures should be employed.

(2) LANDPLANES CONVERTED TO SEAPLANES. A special problem is encountered in the conversion of landplanes to seaplanes. In general, landplanes do not receive corrosion proofing to the same extent as do seaplanes manufactured as such. Corrosion-proofing standards for landplanes converted to seaplanes are divided into two classes: necessary minimum precautions, and recommended precautions.

(i) NECESSARY MINIMUM PRECAUTIONS. The necessary minimum precautions are the minimum corrosion-prevention precautions which must be performed on the initial conversion of landplanes to seaplanes if the landplanes do not already incorporate the precautions mentioned. These procedures are considered the minimum which must be performed in order to safeguard the airworthiness of the converted aircraft and are not in themselves intended to maintain airworthiness for an indefinite period.

(*a*) Unless already protected, exposed fittings or fittings which can be reached through inspection openings should be covered with two coats of zinc chromate primer, paralketone, heavy grease, or similar materials. This applies to items such as wing-root fittings, wing-strut fittings, control-surface hinges, horns, mating edges of fittings and attach bolts, etc.

(*b*) Nonstainless control cables should be coated with grease or paralketone or other similar protective coating, if not replaced with corrosion-resistant cables.

(*c*) Inspection should be made of all readily accessible sections of the aircraft structure including inspection by means of existing inspection openings. Structural parts showing corrosion should be cleaned and refinished if corrosion attack is superficial. If part is severely corroded, it should be replaced with adequately corrosion-proofed parts.

(ii) RECOMMENDED PRECAUTIONS. The recommended precautions are those which are suggested as a means of maintaining such aircraft in condition for safe operation over extended periods of time.

(*a*) Provision of additional inspection openings should be made to assist in detecting corrosion. Experience has shown openings to allow inspection of lower and rearward portion of the fuselage to be particularly desirable.

(*b*) Additional provision for free drainage and ventilation of all interiors should be made to prevent collection of moisture (scoop-type drain grommets).

(*c*) Protection for the interiors of structural steel tubing should be provided. This may be done by air-and-watertight sealing or by flushing with hot linseed oil and plugging openings. Tubing should be inspected for missing sealing screws and the presence of entrapped water. Inspection should also be made for local corrosion around sealing screws, welded clusters and bolted fittings, which may be indicative of entrapped moisture.

(*d*) The fabric of fabric-covered aircraft should be slit longitudinally on the bottom of the fuselage and tail structure for access to these sections. The lower structural members should be coated with zinc chromate primer (two coats), followed by dope-proof paint or wrapping with cellophane tape and rejoining the fabric. This precaution is advisable within a few months after start of operation as a seaplane.

(*e*) Interiors of metal-covered wings and fuselages should be sprayed or fog-sprayed with an adherent corrosion inhibitor.

(*f*) Bags of potassium or sodium dichromate should be placed in the bottom of floats and boat hulls to inhibit corrosion.

(*g*) Exterior surfaces of seaplanes should be washed with clear fresh water immediately following extended water operation, or at least once a day when operated in salty or brackish water. Interior surfaces of seaplanes exposed to spray should also be washed, taking care to prevent damage to electrical circuits or other items subject to injury.

(*h*) Openings into the wings, fuselage, and control-surface members, such as tail-wheel wells, openings for control cables, etc., should be sealed as completely as possible to prevent entry of water.

(d) CLEANERS AND PAINT REMOVERS. It is of importance that aircraft be kept thoroughly clean of deposits of contaminating substances such as oil, grease, dirt, oxide and other foreign materials. The presence of such substances in any appreciable amount constitutes a potential hazard.

(1) MATERIALS. To avoid damage to aircraft through the use of harmful cleaning or paint removing materials only those compounds which conform to existing government specifications or which are products that have been specifically recommended by the aircraft or product manufacturer as being satisfactory for the intended application should be used.

In certain instances a purchaser contemplating the use of removers or cleaners may also establish a specification of his own and have tests made by a testing laboratory of the product being considered in order to insure that it is satisfactory.

(2) REMOVAL OF CORROSION DEPOSITS FROM ALUMINUM. After removing the paint and primers from the surfaces of both bare and clad aluminum alloys the corroded area should be washed with clean fresh water. Corrosion products such at metal flakes, scale, powder and salt deposits may be removed by using fine emery, wire brush or sandpaper. The loose particles may subsequently be washed off with thinner or mineral spirits. In all cases the minimum treatment to produce a smooth clean surface should be used.

(3) SURFACE STAINS. To remove surface stain and superficial etching of metals use a 5-percent-by-weight water solution of sodium bichromate or potassium dichromate. This solution should be allowed to dry on the surface after which excess chromate crystals must be brushed off and the surface subsequently primed and painted.

(4) LIGHT DUTY CLEANERS. Removal of dirt and grease from the surfaces of external parts of the aircraft may be

accomplished by ordinary washing with water. Let the surface dry before further attempting to clean obstinate spots.

One part aircraft cleaning compound, AAF Specification No. 20015, or Navy Specification No. C–147, with 20 parts water may be used on any aircraft, but it is most practicable on fabric covered aircraft where the mixture is applied with a rag or mop, then wiped off with a dry rag. The water mixture may be sprayed on, allowed to remain for 5 to 10 minutes, and then hosed down with water.

(5) REMOVAL OF SPILLED BATTERY ACID. For the neutralization of spilled battery acid use sodium bicarbonate (baking soda), or sodium borate (borax) dissolved in water. The alkali salt should be removed completely after neutralization with copious quantities of water to prevent corrosion.

18.20–7 IDENTIFICATION AND INSPECTION OF MATERIAL. ***(CAA policies which apply to section 18.20.)***

(a) IDENTIFICATION OF STEEL STOCK. The Society of Automotive Engineers (S.A.E.) and the American Iron and Steel Institute (A.I.S.I.) use a numerical index system to identify the composition of various steels. The numbers assigned, in the combined listing of standard steels issued by these groups, represent the type of steel and make it possible to readily identify the principal elements in the material.

The first digit of the four number designation indicates the type to which the steel belongs. Thus "1" indicates a carbon steel, "2" a nickel steel, "3" a nickel chromium steel, etc. In the case of simple alloy steels, the second digit indicates the approximate percentage of the predominant alloying element. The last two digits usually indicate the mean of the range of carbon content. Thus the symbol "1020" indicates a plain carbon steel lacking a principal alloying element and containing an average of 0.20 percent (0.18 to 0.23) carbon. The symbol "2330" indicates a nickel steel of approximately 3 percent (3.25 to 3.75) nickel and an average of 0.30 percent, (0.28 to 0.33) carbon content. The symbol "4130" indicates a chromium-molybdenum steel of approximately 1 percent (0.80 to 1.10) chromium, 0.20 percent (0.15 to 0.25) molybdenum, and 0.30 percent (0.28 to 0.33) carbon. The basic numbers for the four digit series of the carbon and alloy steels may be found in Table 7–1.

(1) INTERCHANGEABILITY OF STEEL TUBING.

(i) "1025" welded tubing as per Specification AN–T–4, and "1025" seamless tubing conforming to Specification AN–WW–T–846, are interchangeable.

(ii) "4130" welded tubing as per Specification AN–T–3, and "4130" seamless tubing conforming to Specification AN–WW–T–850, are interchangeable.

(iii) NE–8630 welded tubing conforming to Specification AN–T–33, and NE–8630 seamless tubing conforming to Specification AN–T–15, are interchangeable.

(b) IDENTIFICATION OF ALUMINUM. To provide a visual means for identifying the various grades of aluminum and aluminum alloys such metals are usually marked with symbols such as Government Specification Number, the temper or condition furnished, or the commercial code marking. Plate and sheet are usually marked with specification numbers or code markings in rows of approximately 5 inches apart. Tubes, bars, rods, and extruded shapes are marked with specification numbers or code markings at intervals of 3 or 5 feet along the length of each piece. The commercial code marking consists of a number which identifies the particular composition of the alloy. A letter preceding the number indicates a modification of the alloy. In addition letter suffixes designate the following:

S—Wrought.

O—Annealed, recrystallized (wrought products only).

H—Strain-hardened. Fractions preceding the letter indicate intermediate tempers as ½H, ¾H, etc.

W—Solution heat treated — unstable temper.

T—Heat-treated and aged to produce stable tempers other than O or H.

RT—Temper resulting from cold working after heat-treatment and aging.

(1) CLAD ALUMINUM ALLOYS. Clad alloys have surface layers of pure aluminum or corrosion-resistant aluminum alloy bonded to the core material to inhibit corrosion. Presence of such a coating may be determined under a magnifying glass by examination of the edge surface which will show three distinct layers.

(2) A TEST FOR DISTINGUISHING HEAT-TREATABLE AND NONHEAT-TREATABLE ALUMINUM ALLOYS. If for any reason the identification mark of the alloy is not on the material, it is possible to distinguish between some heat-treatable alloys and some nonheat-treatable alloys by immersing a sample of the material in a 10-percent solution of caustic soda (sodium hydroxide). Those heat-treatable alloys containing several percent of copper (14S, 17S, 24S) will turn black due to the copper content. High copper alloys when clad will not turn black on the surface, but the edges will turn black at the center of the sheet where the core is exposed. If the alloy does not turn black in the caustic soda solution it is not evidence that the alloy is not heat-treatable, as various high strength heat-treatable alloys are not based primarily on the use of copper as an alloying agent. These include among others, 53S, 61S, 75S, R353, and R361 alloys. Alloys which do not turn black in a caustic soda solution can be identified only by chemical or spectro-analysis as to their composition and heat-treatability.

(c) HARDNESS TESTING. Hardness testing provides a convenient means for determining within reasonable limits the tensile strength of some metals. It has several limitations in that it is not suitable for very soft or very hard metals. In hardness testing the thickness of the specimen being tested and the edge distance should be such that distortion of the metal due to these factors is eliminated. Several readings should be taken and the results averaged. In general, the higher the tensile strength of a metal, the greater is its hardness. Common methods of hardness testing are outlined below.

(1) BRINNELL HARDNESS TEST. In this test a standard load is applied to a smooth surface of metal through a hardened steel ball 1 cm. in diameter. The numerical value of Brinnell hardness is equal to the load divided by the surface area of the resulting spherical impression.

(2) ROCKWELL HARDNESS TEST. In this test a standard minor load is applied to seat a hardened steel ball or a diamond cone in the surface of a metal, followed by the application of a standard major load. The hardness is measured by depth of penetration. Rockwell superficial hardness tests are made using light minor and major loads and a more sensitive system for measuring depth of indentation. It is useful for thinner sections, very small parts, etc.

(3) VICKERS HARDNESS TEST. In this test a small pyramidal diamond is pressed into the metal. The hardness number is the ratio of the load to the surface area of indentation.

(4) TESTING OF STEEL. Hardness testing is a suitable means for determining the tensile properties resulting from the heat treatment of steel. Care should be taken to have case hardened, corroded, pitted, decarburized or otherwise nonuniform surfaces removed to a sufficient depth. Also exercise caution not to cold-work and consequently harden the steel during removal of the surface. The relationship between tensile strength and hardness is indicated in Table 7-2.

(5) TESTING OF ALUMINUM. Hardness tests are useful for testing aluminum alloys chiefly as a means of distinguishing between annealed, cold-worked, heat-treated, and heat-treated and aged material. It is of little value in indicating the strength or quality of heat treatment. Typical hardness values for aluminum alloys are shown in Table 7-3.

(d) INSPECTION OF METALS.

(3) X-RAY OR RADIOGRAPHIC INSPECTION. X-ray may be used on either magnetic or nonmagnetic materials for detecting subsurface voids such as open cracks, blow holes, etc. When a photographic film or plate is used to record the X-ray (in a similar manner to exposing a photographic film), the process is known as radiography. When the X-rays are projected through the part onto a fluorescent screen, the process is known as fluoroscopy. The technique used for radiography should be capable of indicating the presence of defects having a dimension parallel to the X-ray beam of 3 percent of the thickness of the part being radiographed for magnesium alloys and 2 percent for all other metals and alloys. Inspection using a fluoroscopic screen is much less sensitive. Consequently, the radiographic method is usually used for inspection and the fluoroscopic method is used for culling.

Radiographic inspection is extensively used in the aircraft industry for the inspection of all types of casting including sand castings, permanent-mold castings, die castings, etc.. X-ray is particularly useful for this application, since it is capable of disclosing defects which exist below the surface and also since the open types of defects which occur in castings (shrinks, blow holes, dross inclusions, etc.) are readily disclosed by X-rays. In the inspection of forged or wrought metals, on the other hand, X-ray inspection is not used so extensively. This is due to the fact that the process of forging or working causes most defects which originally existed in the metal to become tight-walled cracks which are somewhat difficult to disclose by X-rays.

In case of doubt whether or not X-ray would be suitable for the inspection of certain items, an X-ray laboratory familiar with the examination of aircraft parts should be consulted, as its technicians should be familiar with the limitations of this process.

(4) FLUORESCENT PENETRANT. The trade name for this type of penetrant is Zyglo. In this method of inspection, the article, which may be of metal, plastic material, etc., is first carefully cleaned to permit the fluorescent material to penetrate cracks and defects. It should be noted that cleaning of aluminum may necessitate stripping of any anodizing, since the anodized film, if formed after the defect, could prevent penetration of the fluorescent material. After the article is cleaned, it is either sprayed, painted, or immersed in a bath of fluorescent penetrant. The penetrant is a light oil which has the property of fluorescing or emitting visible light when excited by invisible radiation in the near ultra-violet range (so-called black light). It is important that the penetrant be given sufficient time to penetrate cracks and defects, and for fatigue cracks a minimum of 30 minutes is stipulated by AN–I–30a. Heat may also be applied to facilitate entry of the penetrant. After the penetrant has had sufficient time to enter any defects, the excess on the surface of the article is washed off. This washing should be checked by inspection with black light, by which means any penetrant left on the surface may be detected. After washing, a developer is used to bring out the indication. This developer may be in a liquid form or may be a light powder that absorbs the penetrant as it oozes from cracks and defects in the part. The development may also be aided by application of heat to the part. After the indications have been developed, the part is inspected under black light. Any crevices into which the fluorescent material has penetrated will show as luminous areas.

Indications which appear are usually checked by close inspection with a magnifying glass, by etching with a suitable acid or caustic solution, or it may be necessary to cross-section that part, a procedure which, of course, destroys its usefulness. Usually a skilled operator can determine whether an indication actually shows a defect or whether it is a false indication. Also, the extent of the defect can sometimes be estimated with fair accuracy.

It should be noted that this process of inspection, like all others, has its limitations. If the fluorescent material for any reason is not able to penetrate into a defect, such a defect cannot be detected.

(e) FABRIC. Cotton fabric is used extensively as covering for wing, fuselage, and control surfaces of aircraft. Acceptable grades of fabric for use on civil aircraft may be found in Table 2-1 of CAM 18.20-2. In general, the fabric can be readily identified by a continuous marking to show the manufacturer's name or trademark and specification number. This marking may be found stamped along the selvage. The specification number for Grade "A" fabric is AMS-3806, and for the intermediate grade AMS-3804. The corresponding CAA Technical Standard Order Numbers for these materials are TSO-C-15 and TSO-C-14, respectively.

(f) PLASTICS. Plastics cover a broad field of organic synthetic resins and may be divided into two main classifications—thermoplastic and thermosetting plastics.

(1) THERMOPLASTICS. Thermoplastics may be softened by heat and can be dissolved in various organic solvents. Two kinds of transparent thermoplastic materials are commonly employed in windows, canopies, etc. These materials are known as acrylate base plastics and cellulose acetate base plastics. The acrylate base plastics are manufactured under the commercial names of Plexiglas and Lucite. Cellulose acetate base plastics are manufactured under the commercial trade names of Fibestos, Lumarith, Plastacele, Nixonite, and others. These two base plastics may be distinguished by the absence of color, the greater transparency, and the greater stiffness of the crylates, as compared to the slight yellow tint, lower transparency, and greater flexibility of cellulose acetate.

(2) THERMOSETTING PLASTICS. Thermosetting plastics do not soften appreciably under heat but may char and blister at temperatures of 204 to 260 degrees Centigrade (400 to 500 degrees Fahrenheit). Most of the molded products of synthetic resin composition, such as phenolic, urea-formaldehyde and melamine-formaldehyde resins, belong to the thermosetting group.

18.20-8 AIRCRAFT EQUIPMENT. ***(CAA policies which apply to section 18.20.)***

(a) GENERAL. Aircraft equipment is usually considered as a part of the airplane and many of the maintenance and repair procedures pertinent to the airplane are equally applicable to the various major items of equipment. However, the following general information should be helpful in maintaining the essential equipment items in an airworthy condition.

(b) LANDING GEAR EQUIPMENT.

(1) WHEELS. Wheels should be inspected at periodic intervals for cracks, corrosion, dents, distortion and faulty bearings. In split-type wheels, bolt holes which may have become elongated due to some play in the through-bolt should be reconditioned by the use of ROSAN inserts or other suitable means. The bolts should also be inspected for wear; if excessive wear is evident, the bolts should be replaced. In bolting the wheel halves together, care should be taken to have the nuts properly tightened. These should be inspected periodically to be sure that they are tight so that there will be no movement between the two halves of the wheel. This movement causes elongation of the bolt holes. If the wear is too great to be corrected as stated above, it may necessitate scrapping of the wheel. Grease-retaining felts in the wheel assembly should be kept in a soft, absorbent condition. If they have become hardened they should be cleaned in gasoline; if this fails to soften them, they should be replaced.

(i) CORROSION OF WHEELS. Wheels should be thoroughly cleaned if corroded and then examined for soundness. If bare, corroded spots appear, they should be smoothed up and repainted with a protective coating such as zinc chromate primer and aluminum lacquer or some other equally effective coating to prevent further corrosion. Wheels with severe corrosion which might affect their strength should be replaced.

(ii) DENTED OR DISTORTED WHEELS. Wheels which wobble excessively should be replaced. In questionable cases, the local representative of the Civil Aeronautics Administration should be consulted concerning the airworthiness of the wheels. Dents of a minor nature do not affect the serviceability of a wheel.

(iii) WHEEL BEARINGS. Wheel bearings should be inspected periodically to detect damage caused by maladjustment or foreign material. Damaged or excessively worn parts should be replaced. Bearing cones should be packed with a high melting point grease prior to their installation. When assembling the wheel to the airplane, the axle nut should be tightened just enough to eliminate any drag or wheel side play on the axle.

(2) BRAKES. The clearance between moving and stationary parts of a brake should be maintained in accordance with the manufacturer's recommendations. The brake should be disassembled and inspected periodically and the parts examined for wear, cracks, warpage, corrosion, elongated holes, etc. If any of these or other faults are indicated the affected parts should be repaired, reconditioned, or replaced, depending on the seriousness of the difficulty, in accordance with the manufacturer's recommendations. Surface cracks on the friction surfaces of brake drums occur frequently due to high surface temperature. These may be disregarded as seriously affecting the airworthiness until they become cracks of approximately one inch in length. The brake drums or surfaces then should be replaced.

(i) HYDRAULIC BRAKES. For proper maintenance, the entire hydraulic system from the reservoir to the brakes should be inspected. The fluid in the reservoir should be maintained at the recommended level with the proper brake fluid. Flexible hydraulic hose which has deteriorated due to long periods of service should be replaced. When air is present in the hydraulic system, the system should be bled in accordance with the manufacturer's instructions. Hydraulic piston seal gaskets should be replaced when there is evidence of leakage.

(ii) MECHANICAL BRAKES. The working parts of mechanically operated brakes should be kept free of dirt and foreign matter and should work freely at all times. Excessive play in the linkage system should be kept to a minimum,

(3) TIRES AND TUBES.

(i) TIRES. For maximum safety, it is essential that tires be frequently inspected for cuts, worn spots, bulges on the side walls, and foreign bodies imbedded in the tread and necessary repairs made. It is recommended that repairs be made by reliable local tire repair agencies. Minor cuts may be sealed by filling with commercial tire gum or filler.

(*a*) RETREADING OF TIRES. In retreading new tread may be applied in one of three ways: a top-cap, a full recap, or a retread. In *top-capping,* only a part of the top of the old tread is removed where in *full capping* the work is carried over the shoulders as far as the new tread stock is to be extended. *In retreading,* the old rubber is removed as in the full recapping process except that it is rasped down to the fabric. This removal extends well down over the shoulders. To compensate for the removal of this much rubber, the new tread stock is thicker and has a long bevel on each side which lays over to make new shoulders. In many cases when the original tire manufacturer does the retreading, he replaces the side wall layer of rubber which gives the tire a new appearance. Because of the equipment and skill necessary, recapping work should be given only to local tire agencies which have demonstrated their ability to retread aircraft tires satisfactorily. Additional retreads are permissible after a thorough inspection discloses no side wall bruises, ply separation, or other defects which would indicate that the carcass is not sound enough to justify the retread. More than two retreads are not recommended although there have been cases where a tire gave satisfactory performance after its fourth retread.

(ii) TUBES. Punctured tubes may be repaired by the use of cemented or vulcanized patches. The procedure of making such repairs is substantially identical to that used in connection with automobile tires. The size of the patch should be kept to a minimum and the use of an excessive number of patches, particularly in one area, should be avoided as the weight of the material may contribute to excessive wheel vibration due to tube unbalance. The use of vulcanized patches is recommended because they are considerably more reliable. For such work, a reputable local tire repair agency should be engaged. Reinstalled tires should be inflated, deflated, and again inflated to insure that the inner tube is not pinched. A pinched tube will chafe against the walls of the carcass and a thin spot will result in the rubber. In time, the tube wall will leak at this point. The pinching generally is due to the sticking of the tube to the carcass wall during the first inflation and the failure of the carcass to push out to properly seat against the flange. The tube is then confined to a smaller space and wrinkling (pinching) of the tube results. Complete deflation followed by inflation allows the tube to properly accommodate itself to the carcass which should now seat itself tightly against the flanges.

(4) FLOATS. In order to maintain floats in an airworthy condition, frequent inspection should be made on account of the rapidity with which corrosion takes place on aluminum alloy metal parts, particularly when the aircraft is operated in salt water. Metal floats and all metal parts on wooden floats should be carefully examined for corrosion and corrective action taken in accordance with the procedures described in CAM 18.20–6. Damage to metal floats should be repaired in the general manner as outlined in CAM 18.20–3 (e) pertaining to aluminum and aluminum alloy structures. In the case of wooden floats, repairs should be made in accordance with the general procedure outlined in CAM 18.20–1.

(5) SKIS AND SKI INSTALLATIONS. It is advisable to examine ski installations frequently to keep them maintained in airworthy condition. If shock cord is used to keep the ski runner in proper trim, an examination should be made to see that the cord has enough elasticity to keep the runner in its required attitude and the condition of the cord should be checked to see that it is not becoming loose or badly frayed. If old, weak shock cords are found, they should be replaced. When other means of restraint are provided, they should be examined for excessive wear and binding and replacements should be made when such conditions are found. The points of cable attachment, both on the ski, and the airplane structure, should be examined for bent lugs due to excessive loads having been imposed while taxing over rugged terrain or by trying to break frozen skis loose. If skis which permit attachment to the wheels and tires are used, proper tire pressure should be maintained; underinflated tires may push off the wheels if appreciable side loads are developed in landing.

(i) REPAIR OF SKI RUNNERS. Fractured wooden ski runners usually require replacement. If a split at the rear end of the runner does not exceed 10 percent of the ski length, it may be repaired by attaching one or more wooden cross pieces across the top of the runner using glue and bolts. Bent or torn metal runners may be straightened if minor bending has taken place and minor tears may be repaired in accordance with procedures recommended in CAM 18.20–3 relative to repairs of metal structures.

(ii) SKI PEDESTALS.

(*a*) TUBULAR PEDESTALS. Damaged pedestals made of steel tubing may be repaired by using standard tube splices as shown in Figures 3–2 and 3–12.

(*b*) CAST PEDESTALS. Cast pedestals should not be repaired but should be replaced unless the method of repair is specifically approved by a representative of the Civil Aeronautics Administration.

(*c*) SHEET METAL PEDESTALS. Damaged pedestals made of aluminum alloy sheet should be repaired in the general manner as outlined in CAM 18.20–3(e).

(c) FLARES. Parachute flares are made of materials which are subject to decomposition upon aging. Humidity affects the small igniting charge and also the materials of the candle (illuminant). Hence, the percentage of misfires in old flares is likely to be quite high. To assure unfailing performance of flares, periodic inspection of the flare installation should be made. The inspection should include the entire system starting at the release mechanism in the cockpit and ending at the flare. Such inspection should be attempted only by qualified personnel since inadvertent discharge of such pyrotechnics may cause serious damage. Past experience has indicated that all electrically or pistol-operated flares should be returned to the manufacturer for reconditioning within a maximum period of 3 years and that for mechanically operated flares, this should be done within a maximum period of 4 years.

(d) LIFE PRESERVERS AND LIFE RAFTS.

(1) GENERAL. Inflatable life preservers and life rafts are subject to general deterioration due to aging. Experience has indicated that such equipment may be in need of replacement at the end of 5 years due to porosity of the rubber-coated material. Wear of such equipment is accelerated when stowed on board aircraft because of vibration which causes chafing of the rubberized fabric. This ultimately results in localized leakage. Leakage is also likely to occur where the fabric is folded because sharp corners are formed. When these corners are in contact with the carrying cases, or with adjacent parts of the rubberized fabric, they tend to wear through due to vibration.

In order to keep this equipment in airworthy condition, inspection should be made as follows:

(2) INSPECTION PROCEDURE FOR LIFE PRESERVERS. At 3-month intervals, all life preservers should be inspected for airworthiness. The rubberized material should be examined for cuts, tears, and abrasions and the mouth valves and tubing should be checked for leakage, corrosion, and deterioration. The carbon dioxide cylinder should be removed and the discharge mechanism checked by operating the lever to ascertain that the pin operates freely. The gaskets and valve cores of the cylinder container and the pull cord should be checked for deterioration. If no defects are found, the preserver should be inflated by air to a 2-pound pressure and left for 12 hours. If the preserver still has adequate rigidity at the end of that time, it should be deflated and fitted with CO_2 cylinders having weights not less than that stamped on them by the manufacturer. All cylinders made in accordance with joint Army-Navy Specification No. JAN-C-601 have a minimum permissible weight stamped on them. The use of such CO_2 cylinders is recommended. Prior to repacking the preservers, they should be stamped with the date of inspection as a matter of record.

It is recommended that the above procedure to be repeated every 12-month period, using the CO_2 cartridge for inflation. The carbon dioxide permeates the rubberized fabric at a faster rate than air and will indicate if the porosity of the material is excessive.

(3) REPAIR OF LIFE PRESERVERS. Leaks, as disclosed by immersion in soapy water, should be repaired by the use of patches in accordance with the recommendations of the manufacturer. Corroded metal parts should be cleaned and missing or weakened ties replaced by machine stitching in place as originally done by the manufacturer. Weakened lanyards should also be replaced. Life preservers which do not retain sufficient rigidity after the 12-hour period because of general deterioration and porosity of the fabric are beyond economical repair and should be replaced.

(4) INSPECTION PROCEDURE FOR LIFE RAFTS. Life rafts should be inspected at 3-month intervals for cuts, tears, or other damage to the rubberized material. If the raft is found to be in good condition, the CO_2 bottle(s) should be removed and the raft inflated with air to a 2-pound pressure. The air should be introduced at the fitting normally connected to the CO_2 bottle(s) and the inflated raft left to stand for 24 hours. If the pressure drops below 1¼ pounds, it should be examined for leakage by using soapy water. If the pressure drop is satisfactory, the raft should be considered as being in airworthy condition and returned to service after being fitted with correctly charged CO_2 bottles as determined by weigh-

ing them. It is suggested that the rafts be marked to indicate the date of inspection. Care should be taken to see that all of the raft's required equipment is on board and properly stowed.

At 18-month intervals the above procedure should be repeated, using the CO_2 bottle(s) for inflation. If a single bottle is used for inflating both compartments, it should be noted whether the inflation is proceeding equally to both compartments. Occasionally the formation of "carbon dioxide snow" may occur in one passage of the distributing manifold and divert a larger volume of gas to one compartment which may burst if the mattress valve is not open to relieve the pressure. If the pressure is satisfactory, the raft should be returned to service in accordance with the procedure outlined above.

(5) REPAIR OF LIFE RAFTS. When leaks due to tears, abrasions, or punctures are found, repairs should be made in accordance with the recommendations of the manufacturer. Partially torn away supporting patches on the tube should be recemented so as to restore the raft to its airworthy condition. Mildewed or weak lanyards should be replaced, particularly those by which the CO_2 bottle is operated. This applies also to the line used to attach the raft to the airplane. All metal parts should be checked for corrosion; cleaned, or repaired if found to be defective. If leaky mattress valves have been found, they should be replaced.

(6) RATING. Ratings of military rafts when used in civil aircraft are shown in the following table. Ratings of the 10-, 15-, and 20-man circular rafts built for civil use are stamped on the rafts by the manufacturer.

Type	Capacity			Specification
	Pounds Buoyancy	Men		
		Military	Civil	
A-3 series	1,800	4	6	AF 94-40420
E-2 series	2,500	6	10	AF 94-40618
Mark II	1,000	2	3	Bu Aer M-3R
Mark IV	1,800	4	6	Bu Aer M-3R
Mark VII	2,500	7	10	Bu Aer M-3R

RAFTS MORE THAN 5 YEARS OLD ARE LIKELY TO BE UNAIRWORTHY DUE TO DETERIORATION.

(e) PARACHUTES. With reasonable care, parachutes should last at least 5 years. They should not be carelessly tossed about, left in the airplane so that they may become wet, or left in open places where they may be tampered with. They should not be placed where they may fall on oily floors or be subject to acid fumes from adjacent battery chargers. When repacking is done as required in Part 43 of the Civil Air Regulations, a careful inspection of the parachute should be made by a qualified parachute technician (rigger). If repairs or replacements of parts are necessary to maintain the airworthiness of the parachute assembly, such work should be done by the original parachute manufacturer or by qualified parachute technicians certificated in accordance with Part 25 of the Civil Air Regulations (Parachute Technician's Certificates) or by agencies qualified in accordance with Part 54 (Parachute Loft Certificates and Ratings).

(f) SAFETY BELTS. Safety belts are subject to rapid deterioration due to constant use and effects of aging. Webbing is particularly susceptible to loss of strength because of the effects of high humidity conditions which result in mildewing. The clamping action of the serrations of the commonly used buckle also causes a reduction in the strength of the webbing fibers after long periods of usage. For the above reasons, belts should be periodically replaced, particularly when they are in a frayed condition. It is generally uneconomical for the average airplane owner to have new webbing installed in a belt, particularly if hardware parts also are in need of replacement.

If replacement of webbing or hardware is attempted, however, parts of identical design and material should be obtained. The stitch pattern should at least be identical to the original and the number of threads per inch should be equal to the number used in the original belt. There is no objection to having a greater total length of stitching, provided one line of stitches is not placed over another line. Lines of stitching should be at least $\frac{3}{16}$ inch apart.

A record should be kept, preferably in the log book, stating the extent to which the belt was repaired and the date.

Operators of a fleet of airplanes should also follow the above procedure when repairing or renovating their belts. However, the original belt identification data should be retained so that there will be no question that a type certificated belt had been repaired.

(g) FIRE EXTINGUISHERS. Inspection and maintenance procedure of fire extinguishers should be in accordance with the manufacturer's instructions attached to the extinguisher unit. In connection with carbon tetrachloride extinguishers, owners who desire to refill their own units are cautioned not to refill them with commercial carbon tetrachloride inasmuch as it contains a small percentage of water which ultimately causes corrosion of the moving parts of the extinguisher mechanism.

18.20–9 WINDSHIELDS AND ENCLOSURES. ***(CAA policies which apply to section 18.20.)***

(a) SCOPE. These repairs are applicable to plastic windshields, enclosures, and windows in nonpressurized airplanes. For pressurized airplanes the plastic elements should be replaced, or repaired only in accordance with the manufacturer's recommendations.

(b) TYPES OF PLASTICS. Two types of plastics are commonly used in transparent enclosures of aircraft. These materials are known as acrylate base plastics and cellulose acetate base plastics. The acrylate base plastics are manufactured under such commercial trade names as "Plexiglas", "Lucite", and "Perspex" (British). Cellulose acetate base plastics are manufactured under such commercial names as "Fibestos", "Lumanth", "Plastacele", and "Nixonite".

(1) REPLACEMENT PANELS. Replacement panels should always be of material equivalent to that originally used by the manufacturer of the aircraft since there are many types of transparent plastics on the market. The properties of the different materials vary greatly, particularly in regard to expansion characteristics, brittleness under low temperatures, resistance to discoloration when exposed to sunlight, surface checking, etc. These items have been considered by aircraft manufacturers in selecting materials to be used in their designs and variation or use of substitutes may result in subsequent difficulties.

(c) INSTALLATION PROCEDURES FOR ACRYLIC PLASTICS. In installing a replacement panel, the same mounting method used by the manufacturer of the airplane should be followed. While the actual installation will vary from one type of aircraft to another, the following major principles should be considered in installing any replacement panels.

(1) Never force an acrylic plastic panel out of shape to make it fit a frame. If a replacement panel does not fit easily into the mounting, a new replacement should be obtained or the whole panel heated and reformed. When possible, a new panel should be cut and fit at ordinary room temperatures.

(2) In clamping or bolting acrylic panels into their mountings, do not place the plastic under excessive compressive stress. It is easy to develop more than 1,000 pounds per square inch on the plastic by drawing up a nut and bolt "good and tight." Each nut should be tightened to a firm fit then backed off one full turn.

(3) In bolt installations, spacers, collars, shoulders, or stop-nuts should be used to prevent tightening the bolt excessively. Whenever such devices are used by the airplane manufacturer they should be retained in the replacement installation. It is important that the original number of bolts, complete with washers, spacers, etc., be used.

When rivets are used, adequate spacers or other satisfactory means to prevent excessive tightening of the frame to the plastic should be provided.

(4) Acrylic plastic panels should be mounted between rubber, cork, or other

gaskets to make the installation waterproof, to reduce vibration, and to help to distribute compressive stresses on the plastic.

(5) Acrylic plastics expand and contract approximately three times as much as the metal channels in which they are mounted. Therefore, suitable allowance for dimensional changes with temperature should be made. Clearances of 1/8-inch minimum should be allowed all around the edges of the panel.

(6) In installations involving bolts or rivets, the holes through the plastic could be oversize by 1/8-inch diameter and centered so that the plastic will not bind or crack at the edge of the holes. The use of slotted holes is also recommended.

(7) Panels should be mounted to a sufficient depth in the channel to prevent falling out when it contracts at extremely low temperatures or when the panel is flexed. When the manufacturer's original design permits, panels up to 12 inches long should be mounted to a minimum depth of 1 1/8 inches.

(d) INSTALLATION PROCEDURE FOR CELLULOSE ACETATE PLASTICS. In general, the methods used for installation of cellulose acetate base plastics are similar to those used for acrylic plastics. The coefficient of expansion of cellulose acetate exceeds that of acrylic resins. Also, the cellulose acetate plastics are affected by moisture and will change dimensionally as they absorb water. Therefore, allowance should be made in mounting this type plastic if wide variations in temperature and humidity are to be encountered. As a general rule an allowance of about 1/8-inch per foot of panel length should be made for expansion and 3/16-inch per foot for contraction.

(e) REPAIR OF PLASTICS. Extensively damaged transparent plastic should be replaced rather than repaired whenever possible since even a carefully patched part is not the equal of a new section, either optically or structurally. At the first sign of a crack developing, a hole of 1/8 to 3/16 of an inch in diameter should be drilled at the extreme end of the crack as shown in Figure 9-1. This serves to localize the crack and to prevent further splitting by distributing the strain over a larger area. If the crack is small, stopping it with a drilled hole will usually suffice until replacement or more permanent repair can be made. The following repairs are permissible; however, they should not be located in the line of the pilot's vision in normal flight and landing.

(1) SURFACE PATCH. If a surface patch is to be installed, trim away the damaged area and round all corners. A piece of plastic of sufficient size to cover the damaged area and extend at least 3/4 of an inch on each side of the crack or hole should be cut and the edges beveled as shown in Figure 9-2. If the section to be repaired is curved, the patch should be fitted to the same contour by heating it in an oil bath at a temperature of 248° F. to 302° F. or it may be heated on a hot plate until soft. Boiling water should not be used for heating. The patch should be coated evenly with plastic solvent cement and placed immediately over the hole. A uniform pressure of from 5 to 10 pounds per square inch should be maintained on the patch for a minimum of 3 hours. The patch should be allowed to dry 24 to 36 hours before sanding or polishing is attempted.

(2) PLUG PATCH. In using inserted patches to repair holes in plastic structures, the holes should be trimmed to a perfect circle or oval and the edges beveled slightly. The patch should be slightly thicker than the material being repaired and should have similarly beveled edges. Patches should be installed in accordance with Figures 9-3. The plug is heated until soft and pressed into the hole without cement, and allowed to cool, making a perfect fit. The plug should be removed, the edges cemented and then reinserted in the hole. A firm light pressure should be maintained until the cement has set. Sand or file the edges level with the surface, buff and polish.

(f) CLEANING AND POLISHING TRANSPARENT PLASTIC. Plastics have many advantages over glass for aircraft use, but they lack the surface hardness of glass and care must be exercised while servicing the aircraft to avoid scratching or otherwise damaging the surface.

(1) CLEANING. Plastic should be cleaned by washing with plenty of water and mild soap, using a clean, soft, grit-free cloth, sponge, or bare hands. Do not use gasoline, alcohol, benzene, acetone, carbon tetrachloride, fire extinguisher or de-icing fluids, lacquer thinners, or window cleaning sprays because they will soften the plastic and cause crazing. The plastic should not be rubbed with a *dry* cloth since this is likely to cause scratches and also build up an electrostatic charge which attracts dust particles to the surface. If, after removing dirt and grease, no great amount of scratching is visible, the plastic should be finished with a good grade of commercial wax. The wax should be applied in a thin even coat and brought to a high polish by rubbing lightly with a soft cloth.

(2) BUFFING. If, after removing dirt and grease, the surface is found marred by scratches, hand polish or buff out the scratches. Hand-polishing or buffing should not be attempted until the surface is clean. A soft, open-type cotton or flannel buffing wheel is suggested. Minor scratches may be removed by vigorously rubbing the affected area by hand or with a soft, clean cloth dampened with a mixture of turpentine and chalk or an automobile cleanser applied with a damp cloth. Remove the cleanser and polish with a soft, dry cloth. Acrylate base plastics are thermoplastic and friction created by buffing or polishing too long in one spot can generate sufficient heat to soften the surface. This will produce visual distortion and should be guarded against.

18.20-10 HYDRAULIC SYSTEMS. ***(CAA policies which apply to section 18.20.)***

(a) HYDRAULIC SYSTEMS. Airplane hydraulic systems should be maintained, serviced, and adjusted in accordance with manufacturers' maintenance manuals and pertinent component maintenance manuals. Certain general principles of maintenance and repair which apply are outlined below.

(b) HYDRAULIC LINES AND FITTINGS. All lines and fittings should be carefully inspected at regular intervals to insure airworthiness. Metal lines should be checked for leaks, loose anchorages, scratches, kinks, or other damage. Flexible hose lines should be checked for leaks, cuts, abrasions, soft spots, or other deterioration or damage. Fittings and connections should be inspected for leakage, looseness, cracks, burrs, or other damage. Defective elements should be replaced but may sometimes be repaired.

(1) REPLACEMENT OF METAL LINES. When inspection shows a line to be damaged or defective, the entire line should be replaced. However, if the damaged section is localized, a repair section may be inserted. In replacing lines, always use tubing of the same size and material as the original line. Use the old tubing as a template in bending the new line, unless it is damaged too greatly, in which case a template can be made from soft iron wire. The installation of a new line without bends should be avoided as bends are necessary to allow the tube to expand and contract with temperature changes, and to absorb vibration. Soft aluminum tubing (2SO, 3SO, or 52SO) under 1/4 inch O.D. may be bent by hand. For all other tubing an acceptable tube bending, hand or power, tool should be used. Bending should be done carefully to avoid excessive flattening, kinking, or wrinkling. Minimum bend radii should conform to the values shown in Table 10-1. A small amount of flattening in bends is acceptable but should not exceed an amount such that the small diameter of the flattened portion is less than 75 percent of the original O.D. When installing the replacement tubing it should line up correctly with the mating part and should not be forced into line by means of the coupling nuts.

(2) TUBE CONNECTIONS. Most tubing connections are made using flared tube ends, and standard connection fittings: AN-818 nut and AN-819 sleeve. In forming flares, the tube ends must be cut square, filed smooth, have all burrs and sharp edges re-

moved and be thoroughly cleaned. The tubing is then flared using the correct forming tools for the sizes of tubing and type of fitting. A double flare is used on soft aluminum tubing 3/8 inch O.D. and under, and a single flare on all other tubing. In making the connections hydraulic fluid only should be used as a lubricant. The connection should then be tightened to the proper torque loads as indicated in Table 10-1. This is very important as over-tightening will damage the tube or fitting, and may cause failure, and under-tightening may cause leakage.

(3) REPAIR OF METAL TUBE LINES. Minor dents and scratches in tubing may be repaired. Scratches or nicks no deeper than 10 percent of the wall thickness in *aluminum alloy tubing only*, not in the heel of a bend, may be repaired by burnishing with hand tools. Severe die marks, seams or splits in the tube should not be repaired; such lines should be replaced. Any crack or deformity in a flare is also unacceptable and should be rejected.

A dent less than 20 perent of the tube diameter is not objectionable unless it is in the heel of a bend. Dents may be removed by drawing a bullet of proper size through the tube by means of a length of cable.

A severely damaged line may be repaired, if it is not desired to replace the entire line, by cutting out the damaged section and inserting a repair section consisting of a short section of flared tubing, flaring both ends of the undamaged tube remaining, and connecting with standard unions, sleeves, and tube nuts. If the damaged portion is short enough, the insert tubing may be omitted and the repair made with one union and two sets of connection fittings.

(4) REPLACEMENT OF FLEXIBLE LINES. When replacement of flexible hose lines is necessary, use the same type, size and length hose as the replaced line. If the hose is of the swaged-end type, the entire assembly should be obtained in the correct size for replacement. If it is of the collet, or sleeve, quick-attachable-end type, the assembly should be made up in accordance with the manufacturer's directions, using the prescribed tools, so that the replacement assembly is of the same size and length as the replaced line. The hose should be installed without twisting by keeping the white line on hose straight. Use only hydraulic fluid for lubricating threads.

A hose should never be stretched tight between two fittings as this will result in over-stressing and failure at the ends under pressure. The length of hose should be sufficient to provide about 5 to 8 percent slack.

(c) HYDRAULIC COMPONENTS. Hydraulic components such as pumps, actuating cylinders, selector valves, relief valves, etc., should be disassembled only for maintenance and overhaul in properly equipped shops by qualified personnel. Manufacturer's recommended practices and replacement parts should be used in overhauling such components. If proper servicing facilities are not available, hydraulic equipment in need of repair or overhaul should be replaced by new or overhauled parts and the removed component sent to a qualified agency for overhaul. In making adjustments to such parts as relief valves, pressure regulating valves, etc., the airplane and component manufacturer's instructions should be carefully followed. Hydraulic filter elements should be inspected at frequent intervals and replaced as necessary.

18.20-11 ELECTRICAL SYSTEMS. *(CAA policies which apply to section 18.20.)*

(a) SYSTEMS. All electric equipment, electric assemblies, and wiring installations should be frequently inspected for damage, general condition, and proper functioning to assure the continued satisfactory operation of the electric system. The adjustment, repair, overhaul, and testing of all electric equipment and systems should be accomplished in accordance with the recommendations and procedures set forth in Maintenance Instructions, or manuals published by the Aircraft and Equipment Manufacturers.

(b) PREVENTIVE MAINTENANCE AND OPERATION CHECKING. Frequent visual inspections, operating checks of all electric circuits and equipment, replacement or repair, when deficiencies are found, will go a long way toward the elimination of electrical troubles and hazards in airplanes. A suggested list of things to look for during these inspections and the checks to be performed are itemized below:

- Damaged or overheated equipment and wiring or worn wiring insulation.
- Alignment of electrically driven equipment.
- Poor electric bonding.
- Cleanliness of equipment and connections.
- Proper support of wiring and conduit, and satisfactory attachment to the structure.
- Tightness of connections, terminals and ferrules.
- Continuity of fuses, circuit breakers and wiring.
- Condition of electric lamps.
- Clearance or insulation of exposed terminals.
- Adequacy of safety wire, cotter pins, etc.
- Operational check of electrically operated equipment such as motors, inverters, generators, batteries, lights, etc.
- Voltage check of electric system operation with portable precision voltmeter.

(1) CLEANING AND PRESERVATION. Frequent cleaning of electric equipment to remove dust, dirt, and grime is recommended. Fine emery cloth may be used to clean terminals and mating surfaces if they appear corroded or dirty. Crocus cloth or very fine sandpaper should be used to polish commutators or slip rings. Emery cloth should not be used on commutators since particles from the cloth may cause shorting and burning.

(i) BATTERIES AND BATTERY CONTAINERS. The drain and venting provisions for the battery or battery containers should be checked frequently and if found corroded the compartment and surrounding structure should be washed with a solution of soda and water to neutralize the battery acid.

(2) MISCELLANEOUS CHECK ITEMS. Frequent checks for miscellaneous irregularities such as loose terminal connections, poorly soldered or loose swaged terminals, missing safety wire, loose quick-disconnects, broken wire bundle lacing, broken or inadequate wire clamps, and the clearance of uninsulated terminals that may cause short circuits should be made and replacement or repair accomplished as a part of routine maintenance.

(c) MAJOR ADJUSTMENTS. Major adjustments of items of equipment such as regulators, generators, contractors, control devices, inverters, and relays should be accomplished outside the airplane on the test stand or test bench where all the necessary instruments and equipment are at hand. The adjustment procedures outlined by the equipment manufacturer should be followed.

(d) EQUIPMENT REPLACEMENT. Damaged, worn-out, and defective electric equipment should be replaced with identical items or with equipment equivalent to the original in operating characteristics, mechanical strength, and the ability to withstand the environmental conditions encountered in the operation of the airplane.

(e) WIRING AND TERMINAL REPLACEMENT. When replacing electric wiring great care should be exercised to see that the wires are cut to the correct length, and that upon installation in the airplane, they are routed, supported, protected, and terminated in such a manner that insulation abrasion is kept to a minimum.

(1) ATTACHING TERMINALS TO ELECTRIC CABLE.

(i) GENERAL PROCEDURE. Terminals may be attached to electric cable by swaging, staking, or soldering. The required length of insulation should be stripped from the conductor carefully so that nicked or cut wire strands in no case exceed 10 percent of the total number of strands. Satisfactory terminal attachment is of primary importance and should be accomplished by use of the proper tools and careful insertion of the stripped cable in the terminal. The terminal manufacturers' instructions concerning cable size and the proper tools to be used for each terminal should be followed.

(ii) TYPES OF TERMINALS. In general, the solderless type of swaged or staked terminals, disconnect splices, terminal blocks, and connectors should be used. The use of standard type soldered terminal multi-pin connectors should be confined to equipment where frequent connecting and disconnecting is necessary. Where soldered terminals are necessary they may be cleaned with very fine emery cloth, sandpaper, or alcohol. Proper tinning of the wire strands and terminal lug solder cups are essential prior to soldering and during these operations only pure resin flux should be used.

(iii) TERMINAL INSULATION. Insulating tubing of the vinyl variety should be placed over each cable terminal at splices, terminal blocks, equipment, and all connections of both plug and receptacle of connectors, unless the terminals used are of a preinsulated variety. The tubing should be secured, so as to prevent its slipping from the electric connection.

(iv) REPLACEMENT OF ELECTRIC CABLES. When abraded electric cable is found it should be replaced with new cable of the same or equal quality and if possible the cause of such abrasion should be eliminated by providing additional support or clamping devices. If only a short portion of an electric cable of considerable length appears abraded, the defective portion can be replaced and the new cable connected to the old by means of permanent or quick-disconnect splices. The splices should be protected by insulating tubing.

(2) SELECTION OF ELECTRIC CABLE.

(i) TYPE OF CABLE. Electric cables meeting the performance requirements of Specification AN–J–C–48 (copper) and AN–C–161 (aluminum) are satisfactory for use in certificated aircraft.

(ii) CRITERIA FOR SELECTION. The two criteria upon which the selection of electric cable size should be based, when considering an alteration, are current carrying capacity and voltage drop.

(*a*) The selected cable should not carry continuously or intermittently current in excess of the ampere values indicated by Curves 1, 2, and 3 on Figure 11–1.

(*b*) The voltage drop in the main power cables from the generation sources or the battery to the bus should not exceed two percent (2.0%) of the regulated voltage, when the generator is carrying rated current and the battery is being discharged at the 5-minute rate.

(*c*) The voltage drop in the load circuits, between the bus and the utilization equipment, should not exceed the values shown in the following tabulation:

Nominal system voltage	Allowable voltage drop	
	Continuous operation	Intermittent operation
14	0.5	1
28	1	2
115	4	8
200	7	14

(iii) ELECTRIC CABLE CHART. This chart, Figure 11–1, applies to cable carrying direct current and is based on copper conductor cable manufactured in accordance with Specification AN–J–C–48, whose current ratings are given in Specification AN–W–14. Curves 1, 2, and 3 thereon intersect the vertical cable size lines at the maximum ampere rating for the specified conditions indicated on the chart.

(*a*) EXAMPLES OF HOW TO USE THE ELECTRIC CABLE CHART—Figure 11–1.

(*1*) KNOWING THE CABLE LENGTH AND AMPERE LOAD. Determine the required cable size so as not to exceed one volt drop as follows: Select the cable length from the scale at the left and follow it horizontally across the chart to the right until it intersects the required diagonal ampere line. Then read the cable size on the nearest or preferably the nearest vertical cable size line to the right.

Example: Measured cable length 50 feet, continuous current 25 amperes — determine cable size. From the left scale follow horizontal line 50 across chart to the right until it intersects the diagonal 25-ampere line. The 25-ampere line is slightly more than midway between the 20- and 30-ampere lines as the scale is logarithmic. The vertical cable size line to the right of this intersection is numbered 8, and therefore a No. 8 cable size will be needed.

(2) KNOWING THE CABLE SIZE AND AMPERE LOAD. Determine the maximum cable length so as not to exceed one volt drop as follows: Select the cable size from the scale at the bottom of the chart and follow the vertical cable size line until it intersects the required diagonal ampere line. Then read the maximum distance in feet, that the cable can be run, by horizontally projecting the point of intersection to the scale at the left.

Example: Cable size No. 2, continuous current 150 amperes — determine maximum cable length in feet. From the bottom scale follow the No. 2 vertical cable size line until it intersects the diagonal 150-ampere line. Projecting this point horizontally to the scale at the left it is determined that 38 feet is the maximum distance that the No. 2 cable carrying 150 amperes can be run without exceeding one-volt drop. It should be noted, however, that the point of intersection falls below Curve 1 and if the cable is to be installed in a close fitting conduit or even a large bundle it would be preferable to use a No. 1 or No. 1/0 cable, depending on the known factors of the installation. Naturally the maximum distance that these larger cables can be run without exceeding one-volt drop will also be greater than that previously determined for the No. 2 cable.

(3) FOR OTHER THAN ONE-VOLT DROP:

Examples: Determine cable size for various voltage drops, measured cable length 100 feet, continuous current 20 amperes; also determine maximum cable lengths in feet for various voltage drops, using cable size No. 10, continuous current 20 amperes. *(See tables at bottom of page.)*

(iv) RESISTANCE. The resistance of the current return path through the aircraft structure is always considered negligible. However, this is based on the assumption that adequate bonding of the structure or

Voltage drop	Enter chart (feet)	Amperes	Cable size from chart	Check
1	100	20	No. 6	VD = (.000436)[b] (100) (20) = .872
0.5	$\frac{100}{0.5} = 200$	20	No. 4	VD = (.000274)[b] (100) (20) = .548
4	$\frac{100}{4} = 25$	20	No. 12	VD = (.00188)[b] (100) (20) = 3.76
7	$\frac{100}{7} = 14$	20	No. 14[a]	VD = (.00299)[b] (100) (20) = 5.98

[a] It should be noted that the No. 14 cable should not be used if a considerable portion of its 100-foot length is to be confined in conduit, large bundles, or locations of high ambient temperature, as the intersection of the cable size and current lines falls below Curve 1.
[b] Resistance values from Table 11–1.

Voltage drop	Wire size	Amperes	Max. length (feet) from chart at voltage drop indicated	Check
1	No. 10	20	45	VD = (.0011)[a] (20) (45) = .990
0.5	—	—	(45) (.5) = 22.5	VD = (.0011)[a] (20) (45) = .495
4	—	—	(45) (4) = 180	VD = (.0011)[a] (20) (45) = 3.96
7	—	—	(45) (7) = 315	VD = (.0011)[a] (20) (315) = 6.93

[a] Resistance values from Table 11–1.

a special electric current return path has been provided which is capable of carrying the required electric current with a negligible voltage drop. The measured resistance from the ground point of a generator or the battery to the ground terminal of any electric device should not exceed 0.005 ohm.

(v) ALUMINUM CABLE. From Table 11-1 it will be noted that the conductor resistance of aluminum cable and that of copper cable two numbers higher are practically identical. Accordingly, the Electric Cable Chart, Figure 11-1, can be used when it is desired to substitute aluminum cable and the proper size can be selected by remembering to reduce the copper cable size by two numbers or by referring to Table 11-1. The use of aluminum cable sizes smaller than No. 8 is not recommended.

(f) ELECTRIC BONDING. Bonding is the electrical interconnecting of the metal parts of the airplane in such a manner as to produce low-resistance paths for electric currents. The purpose of bonding is to:

Reduce interference to radio equipment.
Reduce fire hazard.
Provide a return path for electrical current in single-wire electrical systems.
Prevent control hinges from being welded together by lightening strikes.

Bonding is accomplished by using clamps or flexible wire (pigtails) or both to interconnect the metal parts of the airplane. Fuel and oil lines are usually most easily bonded to structure at each usual point of attachment to structure by using clamps having a metal bonding strip in them or by using a flat strip of thin aluminum or plated copper between the line and the clamp. Control and brace wires usually require pigtails. The pigtails should be as short as practicable and should be attached to the control cables bracing by clamping which does not tend to flatten the wire. Surfaces to which the bonding attaches should be free from dirt, paint or other nonconducting material. Pigtails should not be soldered to control or brace wires.

18.20-12 INSTRUMENTS. ***(CAA policies which apply to section 18.20.)***

(a) GENERAL.

(1) INSTRUMENT INSTALLATION AND MAINTENANCE. Care should be taken with instruments to prevent their accidental damage.

When instruments do not give proper indications, they should be sent to an approved instrument overhaul and repair station or returned to the manufacturer for servicing.

(2) VIBRATION INSULATION. Instruments should not be subjected to excessive vibrations. When shock-insulated panels are used, the mountings should be periodically checked for condition and the panels for alignment. When necessary to replace shock mounts, units of the same characteristics should be used. Only flexible connector tubing should be used to join the end of lines to the instruments. Care should be exercised to insure that the instrument panel does not contact any parts of the airframe when vibrating normally.

(b) PITOT-STATIC SYSTEM.

(1) SYSTEM COMPONENTS. The pitot-static system consists of the pitot and static pressure pickups, lines, tubing, water drains and traps, and the various pressure indicators such as the altimeter, air-speed and rate of climb indicators connected to the system.

(i) PITOT-STATIC TUBE. If the pitot-static tube has been damaged beyond repair it should be replaced. The tube should have its axis parallel to the longitudinal axis of the aircraft when in cruising flight configuration. All repairs and alterations on the pitot-static system should be made in conformance with the manufacturers' recommendations.

(ii) HEATER NOT OPERATIVE. If the heating element of the pitot-static becomes inoperative, it should be replaced. In the types of tubes where the element is not replaceable, the complete tube should be replaced. The voltage at the heater terminals should not be less than 85 percent of the rated system voltage.

(iii) CLOGGING OF PITOT-STATIC TUBE DRAINS. If water has entered the system, the drains in the pitot-static head should be probed with a fine wire to remove dirt or other obstructions. The bottom static openings act as drains for the static chamber and these holes should be checked at regular intervals to preclude malfunctioning of the system.

(iv) RELOCATION OF PITOT-STATIC TUBE. If relocation of the pitot-static tube is necessary it should be done with due consideration of the following:

(*a*) Freedom from aerodynamic disturbances caused by the aircraft.

(*b*) Location protected from accidental damage.

(*c*) Alignment with the longitudinal axis of the aircraft when in cruising flight configuration.

(*d*) Any alteration should best be made in conformance with the manufacturers' recommendations.

(2) PITOT-STATIC LINES.

(i) POOR DRAINAGE OF LINES. If drainage is poor, check the line diameter. If the diameter is less than 3/8 inch O.D., it should be replaced with 3/8 inch O.D. diameter lines to overcome the difficulty, as water will not drain freely in smaller size lines.

(ii) REPLACING THE LINES. If necessary to replace lines, the following installation practices should be observed:

(*a*) Attach lines to airframe at regular intervals by means of clamps.

(*b*) Do not clamp lines at end fittings.

(*c*) Maintain slope of lines toward drains so that proper drainage will be effected.

(*d*) Use thread lubricant on fittings, preventing excess lubricant from entering lines.

(*e*) Check the lines for leaks.

(iii) LEAK TESTING STATIC LINES. The static line openings should be connected into a common line to which a manometer or a reliable pressure gauge and a suction source are connected. Apply suction equivalent to 1,000 feet altitude (1.05 inches of mercury or 14.24 inches of water) and hold. In 1 minute, the suction should not lose more than the equivalent of 150 feet of altitude (approximately 2.18 inches of water). WARNING: DO NOT APPLY POSITIVE PRESSURE TO THE STATIC LINES.

(iv) LEAK TESTING — PITOT LINES. Seal drain holes, and connect pitot pressure openings to a tee to which a source of pressure and a manometer or reliable airspeed indicator is connected. Apply pressure to cause airspeed to indicate 150 miles per hour (0.82 inches of mercury or 11.18 inches of water positive pressure). Pinch off the pressure source, and the pressure after one minute of time should not indicate less than 140 miles per hour. WARNING: DO NOT APPLY SUCTION TO PITOT LINES.

(v) MAINTENANCE OF LINES. Inspection of the lines should be made periodically. Water accumulation can be removed by opening the drain cocks. If the installation is not properly self-drained, disconnect the lines from the instruments and carefully "blow" the lines with clean dry air.

(c) MAGNETIC DIRECTION INDICATOR (compass).

(1) CORRECTION FOR ERRORS IN MAGNETIC DIRECTION INDICATOR.

(i) SWINGING THE COMPASS (ground). When the magnetic direction indicator does not yield satisfactory directional indications, it can be calibrated by the "ground swinging" technique as follows:

(*a*) Remove aircraft to location free from influence of steel structures, underground pipes and cables, reinforced concrete, or other aircraft.

(*b*) Place the aircraft in level flying position.

(*c*) Remove compensating magnets from chambers or reset the fixed compensating magnets to neutral positions, whichever is applicable, before swinging.

(*d*) Check compass for fluid level and

cleanliness. If fluid is required, it should be added before compensation.

(*e*) Check the pivot friction of compass by deflecting the card with a small magnet. The card should rotate freely in a horizontal plane.

(*f*) If radio is used in aircraft, there should be corrections noted for "Radio On" and "Radio Off" conditions.

(*g*) Align the aircraft with the North magnetic heading and compensate with compensating magnets. Repeat for the East magnetic heading. Then place on South and West magnetic headings and remove half of indicated error by adjusting compensators. Engine(s) should be running.

(*h*) Swing the aircraft on successive 30° headings through 360°. Placard should be marked to indicate correction at each 30° heading showing "Radio On" and "Radio Off" corrections.

(ii) INDICATOR CANNOT BE PROPERLY COMPENSATED. The pilot's indicator should have deviation of less than 10° at any heading. When this maximum is exceeded, a new location for the indicator should be considered, unless the condition causing the error can be removed permanently.

(iii) ERRATIC INDICATIONS OF MAGNETIC INDICATOR. If severe deviations are encountered, they may be due to iron or steel items being carried in the aircraft, and located too close to the magnetic direction indicator. Caution must be taken to properly locate articles of this nature away from the vicinity of the indicator.

18.20-13 ENGINES AND FUEL SYSTEMS. ***(CAA policies which apply to section 18.20.)***

(a) ENGINES, GENERAL. In repairing or overhauling aircraft engines, all repair agencies should be guided by the recommendations and procedures set forth in the respective instruction books, manuals, or service bulletins for the installation, inspection, and maintenance of aircraft engines, published by the aircraft engine manufacturers for each type of engine. The overhaul period for aircraft engines used in nonscheduled service should be determined from the manufacturers' general recommendations in this respect and the condition of each engine involved.

(1) ROTATING, RECIPROCAATING, AND HIGHLY STRESSED ENGINE PARTS. The rotating, reciprocating, and highly stressed parts of all aircraft engines should be subjected to a critical inspection at the time of overhaul. Whenever possible, this inspection should be supplemented by a wet or dry magnetic dust inspection of the steel parts. In such a case, a copy of the report covering the findings of the magnetic dust inspection should be appended to the original Repair and Alteration Form in the case of a major repair.

(2) REBUILT ENGINES. A rebuilt engine is defined as a used engine which has been completely disassembled, inspected, repaired as necessary, reassembled, tested, and approved in the same manner and to the same tolerances and limits as a new engine. Component parts of such engines may be either used parts or new parts. The used parts may be either the parts from the same engine or from other service engines, but they must conform to production drawing tolerances and limits to which new parts must conform. In addition, all parts, either new or used, meeting approved oversize and undersize dimensions acceptable for new engines, are also eligible.

(3) CRANKSHAFTS. Crankshafts should be carefully inspected for misalignment and if bent beyond the manufacturer's permissible limit for service use, should not be repaired, but should be replaced. Worn journals may be repaired by regrinding in accordance with the manufacturer's instructions. If the original fillets are altered at any time, their radii should not be reduced and their surfaces should be polished free of all tool marks.

(4) REPLACEMENT PARTS IN CERTIFICATED ENGINES. Only engine parts which are approved by the Civil Aeronautics Administration should be used in making replacements in certificated aircraft engines.

(5) RUN-IN TIME. After an aircraft engine has been overhauled, it should be run-in in accordance with the pertinent aircraft engine manufacturer's instructions. If no special test stand, test club, and other equipment are available, the engine may be run-in on the aircraft and the aircraft should be headed into the wind during the run-in on the ground so that the maximum cooling effect will be obtained. Proper cooling during run-in cannot be overemphasized. The manufacturer's recommendations concerning engine temperatures and other criteria should be carefully observed.

(6) RE-USE OF SAFETYING DEVICES. Cotter pins and safety wire should never be used a second time. Flat steel-type wrist-pin retainers and thin lock washers likewise should be replaced, but special coil spring or plugtype retainers need not be replaced at overhaul if the manufacturer's recommendations permit re-use.

(7) SELF-LOCKING NUTS FOR AIRCRAFT ENGINES AND ACCESSORIES. Self-locking nuts may be used on aircraft engines provided the following criteria are met:

(i) Where their use is specified by the engine manufacturer in his assembly drawing parts list, and bills of material which are approved by the Civil Aeronautics Administration.

(ii) When the nuts will not fall inside of engine should they loosen and come off.

(iii) When there is at least one full thread protruding beyond the nut.

(iv) If cotter pin or locking-wiring holes are in the bolt or stud, the edges of these holes should be well-rounded to preclude damage to the lock nut.

(v) The effectiveness of the self-locking feature should be checked and found to be satisfactory prior to its re-use.

(vi) Engine accessories should be attached to the engine by means of the types of nuts furnished with the engine. On many engines, however, self-locking nuts are furnished for such use by the engine manufacturer for all accessories except the heaviest, such as starters and generators.

(vii) On many engines, the cylinder baffles, rocker box covers, drive covers and pads, and accessory and supercharger housings, are fastened with fiber insert lock nuts which are limited to a maximum temperature of 250° F. inasmuch as above this temperature the fiber will usually char and consequently lose its locking characteristic. On locations such as the exhaust-pipe attachment to the cylinder, a lock nut which has good locking features at elevated temperatures will give invaluable service. In a few instances, fiber insert lock nuts have been approved for use on cylinder hold-down studs. This practice is not generally recommended since (1) especially tight stud fits to the crankcase must be provided, and (2) extremely good cooling must prevail so that low temperatures exist at this location on the specific engines for which such use is approved.

(viii) It is necessary that all proposed applications of new types of lock nuts or new applications of currently used self-locking nuts be investigated adequately since most engines require some specially designed nuts. Such specially designed nuts are usually required for one or more of the following reasons:

(a) to provide heat resistance;

(b) to provide adequate clearance for installation and removal;

(c) to provide for the required degrees of tightening, or, locking ability which sometimes requires a stronger, specially heat-treated material, a heavier cross-section, or a special locking means;

(d) to provide ample bearing area under the nut to reduce unit loading on softer metals;

(e) to prevent loosening of studs when nuts are removed. Information concerning approved self-locking nuts and their use on specific engines is usually found in engine manufacturers' manuals or bulletins. If the desired information is not available, it is suggested that the engine manufacturer be contacted.

(b) WELDING IN THE REPAIR OF ENGINES.

(1) GENERAL. In general, welding of highly stressed engine parts is not recommended. However, under the conditions given below, welding may be accomplished if it can be reasonably expected that the welded repair will not adversely affect the airworthiness of the engine:

(i) when the weld is externally situated and can be inspected easily;

(ii) when the part has been cracked or broken as the result of unusual loads not encountered in normal operation;

(iii) when a new replacement part of obsolete-type engine is not available;

(iv) when the welder's experience and equipment employed will insure a first quality weld in the type of material to be repaired and will insure restoration of the original heat treat in heat-treated parts. Also refer to CAM 18.20–3(b) for information on process details.

(2) WELDING OF MINOR PARTS. Many minor parts not subject to high stresses may be safely repaired by welding. Mounting lugs, cowl lugs on cylinders, covers, etc., are in this category. The welded part should be suitably stress-relieved after welding.

(c) METALLIZING. Metallizing should not be done on any internal part of an aircraft engine except when it is proved conclusively to the Civil Aeronautics Administration that the metallized part will not adversely affect the airworthiness of the engine. Metallizing the finned surfaces of steel cylinder barrels with aluminum may be accomplished since many engines are originally manufactured in this manner.

(d) PLATING.

(1) GENERAL. Plating may be restored on an engine part when accomplished in accordance with the manufacturer's instructions.

(2) PLATING OF HIGHLY STRESSED PARTS. In general, chromium plating should not be applied to highly stressed engine parts. Certain applications of this nature have been found to be satisfactory. However, the processes to be used should be approved in all details by the Civil Aeronautics Administration. Porous chromium-plated cylinder walls have been found to be satisfactory for practically all types of engines. Dense or smooth chromium plating with unroughened surfaces, on the other hand, has not been found to be generally satisfactory. Dense chromium plating of the crankpin and main journals of small engine crankshafts has been found to be satisfactory except where the particular crankshaft is already of marginal strength.

Refer to CAM 18.20–6(b) (2) for further information on plating.

(3) PLATING OF MINOR PARTS. Plating, including chromium plating, may be utilized to restore worn low-stressed parts of an engine, such as accessory drive shafts and splines, propeller shaft ends, and the seating surfaces for roller- and ball-type bearing races.

(e) CORROSION PREVENTION. The use of strong solutions which contain strong caustic compounds and of all solutions, polishers, cleaners, abrasives, etc., which have an adverse effect on the airworthiness of the engine due to corrosive action will not be considered satisfactory.

Refer to CAM 18.20–6 for further details.

(f) ENGINE ACCESSORIES. The engine accessories should be overhauled and repaired in accordance with the recommendations of the engine manufacturer and the accessory manufacturer.

(g) FUEL TANKS AND FUEL SYSTEMS.

(1) WELDED OR RIVETED TANKS. If tanks are made of commercially pure aluminum, 3S, 52S, or similar alloys, they may be repaired by welding. Tanks made from heat-treatable aluminum alloys are generally assembled by riveting. In case it is necessary to rivet a new piece in place, the patch should be of the same material as the tank, and a sealing compound that is insoluble in gasoline should be employed in the seams. Bakelite varnish, "Glyptalac," marketed by General Electric Company, "Thiokol" made by the Thiokol Corporation, Yardville, N. J., or "Neoprene" made by the

E. I. duPont de Nemours and Company, zinc chromate compound (type No. 2) made by W. P. Fuller Company, of Los Angeles, Calif., are examples of sealing compounds which are acceptable when nonaromatic fuels are used. If aromatic fuels are used, special sealing compounds which are resistant to aromatic fuels should be employed.

(i) REMOVAL OF FLUX AFTER WELDING. It is especially important after repair by welding, to completely remove all flux in order to avoid possible corrosion. Therefore, promptly upon completion of welding, the tank should be washed both inside and outside with liberal quantities of hot water, and drained. Next, immerse it in either 5 percent nitric or 5 percent sulphuric acid, or fill the tank with this solution (in which case also wash the outside with the same solution). Permit this acid to remain in contact with the weld about 1 hour and then rinse thoroughly with clean fresh water. The efficiency of the cleaning operation may be tested by applying some acidified 5 percent silver nitrate solution to a small quantity of the rinse water that has been used to last wash the weld. If a heavy white precipitate is formed, the cleansing has been insufficient and the washing should be repeated.

(2) FUEL TANK CAPS, VENTS, AND OVERFLOW LINES. Fuel tank caps should be inspected as to the integrity of the gasket, and vents should be inspected to ascertain that they are clear. Overflow lines should be inspected to ascertain that the integrity of the material and connections are satisfactory. Care should also be taken to ascertain that the vent exit is in proper position.

(3) FUEL LINES. Aluminum or aluminum alloy tubing should not be annealed after forming or at overhaul periods as is required practice with copper tubing. Fuel lines should be thoroughly inspected as to integrity of end fittings, bends, or kinks beyond recommended bend radii, for foreign material within the lines, and for integrity of the material such as would be affected by abrasion, acid, heat, or swelling in the case of rubber impregnated lines. Too sharp bends, kinks, evidence of excessive heat, abrasion, or a change in the material is cause for replacement.

(4) FUEL STRAINERS AND SEDIMENT BOWLS. The adjusting nut located at the bottom of the bowl of the fuel strainer should be positively safetied in position. This nut should be tightened only with the fingers. If leakage still occurs, do not tighten with pliers but replace the cork gasket between the glass bowl and the screen. The screens of all strainers should be periodically inspected for foreign material or rupture. Screens should only be replaced by those recommended by the manufacturer as the mesh size affects the fuel flow through them. Sediment bowls shall be given frequent inspections for water or solid material.

18.20–14 PROPELLERS. *(CAA policies which apply to section 18.20.)*

(a) MAINTENANCE OF PROPELLERS.

(1) GENERAL. The propeller is easily accessible for visual inspection and should always be checked before a flight to determine that no damage has occurred.

(i) WARNING STRIPES. Many persons have been fatally injured by walking into whirling propellers. Painting a warning stripe on the propeller serves to reduce chances of such injuries. Approximately 4 inches of the propeller tips should be covered on both sides with an orange-yellow nonreflecting paint or lacquer. The drain holes in the metal tipping of wood blades should be opened up after the tips have been painted.

(2) WOOD PROPELLERS. Wood propellers are in the same category as any other wood structural part of the airplane. It is continually necessary to ascertain that the glue joints are in good condition and that the finish on the entire propeller will protect the propeller from absorbing moisture. Two-bladed wood propellers should always be left or stored, whether on or off of an airplane, in a horizontal position to prevent unbalance from moisture absorption. When moving an airplane, special care should be exercised to

avoid bumping the propeller. The practice of pushing or pulling on a propeller blade to move an airplane should be avoided; it is extremely easy to impose forces on a blade in excess of those for which the blade is designed.

(i) FIXED-PITCH WOOD PROPELLERS. Loose hub bolts and bolts installed through the lightening holes in the integral hub flange of certain engine crankshafts continue to be the cause of the majority of the serious difficulties experienced with propellers of this type. Either of the conditions, if uncorrected, will ultimately cause the loss of the propeller.

(*a*) LOOSE BOLTS. Loose hub bolts cause elongated bolt holes and damage to the hub bolts. When not corrected, the bolts break off or friction causes enough heat to affect the glue and char the wood. After successive running, checks start at the bolt holes. These checks are caused, or at least accentuated, by shrinkage of the wood due to the excessive heat generated. If allowed to progress, the propeller usually flies apart or catches fire.

(*b*) INCORRECTLY INSTALLED BOLTS. On some engines equipped with a crankshaft having an integral propeller hub flange, the outer edge of the lightening holes is at the same radius as the corresponding edge of the propeller hub bolt holes. When inserting the bolts through the propeller, care must be exercised so that the bolts are inserted through the proper holes in the flange. Cases have been reported where the bolts were inserted through the larger lightening holes and, accordingly, the bolt nuts bore only on the outer edge of the lightening holes. In such cases, continued running of the propeller may cause the bolt heads or nuts to slip off the flange and through the large openings in the flange, resulting in the subsequent loss of the propeller.

(*c*) CORRECTIVE MEASURES. Both of the conditions discussed above are very easy to detect and correct. In case the hub flange is integral with the crankshaft, first ascertain that the bolts are properly installed. Then make the inspection for bolt tightness in the same manner as for any other propeller hub. Use an open end wrench to determine hub bolt tightness and if the nuts can be turned, remove the cotter keys and draw up the nuts to the desired setting. Hub bolts should be tightened, preferably with a torque wrench, to the recommended values which usually range from 15 to 23 foot-pounds. If no torque wrench is available, an ordinary socket wrench may be used. This socket wrench should have a 1-foot extension lever and the wrench pulled up with the recommended force, 12 inches away from the center of the bolt which is being tightened. The tightening is best accomplished by tightening each bolt a little at a time, being sure to tighten alternate bolts which are diametrically opposite. Care should be taken not to overtighten the hub bolts, thereby damaging the wood underneath the hub flanges. The practice of overtightening bolts to draw a propeller into track should definitely be avoided. Safetying of the nuts should be accomplished by means of cotter keys of the proper size, or heavy safety wire twisted between each nut. A continuous length of single safety wire should never be used as wire failure will result in all nuts becoming unsafetied.

(*d*) INSPECTION ON NEW INSTALLATIONS. On new propeller installations the hub bolts should always be inspected for tightness after the first flight and after the first 25 hours of flying. Thereafter, the bolts should be inspected and checked for tightness at least every 50 hours. No definite time interval can be specified, since bolt tightness is affected by changes in the wood caused by the moisture content in the air where the airplane is flown and stored. During wet weather, some moisture is apt to enter the propeller wood through the drilled holes in the hub. The wood swells but, since expansion is limited by the bolts extending between the two flanges, some of the wood fibres are crushed. Later, when the propeller dries out during dry weather or due to heat from the engine, a certain amount of propeller hub shrinkage takes place and the wood no longer completely fills the space between the two hub flanges. Accordingly, the hub bolts become loose.

(*e*) INSPECTION ON OLD INSTALLATIONS. The propeller should be removed from the engine at engine overhaul periods. Whenever the propeller is removed, it should be visually inspected on the rear surface for any indication of cracks in the wood or in the metal hub. When any indications are found, the metal hub should be disassembled from the propeller. The hub should then be inspected by the magnetic particle process for cracks, especially in the flange and keyway. The bolts should be inspected for wear and cracks at the head and threads and, if cracked or worn, should be replaced with new AN bolts. The propeller should be inspected for elongated bolt holes, enlarged hub bore, and checks or cracks inside of bore or anywhere on the propeller. Propellers found with any of these defects should not be used until repaired. A method for repairing these defects is outlined in CAM 18.20–14(e).

(*f*) FINISH, TRACK, AND BALANCE. Touch up with varnish all places where the finish is worn thin, scratched, or nicked. Track and balance the propeller, and coat the hub bore and bolt holes with some moisture preventive such as asphalt varnish. In case the hub flange is integral with the crankshaft of the engine, final track should be made after the propeller is installed on the engine. In all cases where a separate metal hub is used, final balance and track should be accomplished with the hub installed in the propeller.

(ii) DETACHABLE WOOD OR COMPOSITION BLADE PROPELLERS. In general there are two methods of attaching blades of this type: (*a*) the lag screw, and (*b*) the spun-on metal shank covering. It is very important to protect the shank section from moisture changes to prevent swelling and subsequent loosening. A good precaution is to cover the propeller with a well-fitting waterproof cover when not in use.

In the case of Item (*a*), tightness of the lag screws should be inspected according to the manufacturer's instructions. If any lag screws are found to be broken, the matter should be taken up with the manufacturer and a CAA representative should be notified.

In the case of Item (*b*), a slight lateral movement is sometimes noticeable in blades fitted with spun-on metal shank coverings. This looseness may result from a desire on the part of the manufacturer to insure that the wood fibers at the outer end of the metal sleeve are not crushed during the spinning-on process. Any such slight looseness is not considered unairworthy unless there are indications that the blade has moved outward. Blades of this type, therefore, should be examined at the section where the wood blade enters the metal shank covering, for any indication that the blade has moved outward under the action of centrifugal force. A movement of this kind indicates that the wood shank has been affected by moisture since manufacture. Any blade found to have moved outward should be referred to the manufacturer.

(3) METAL PROPELLERS. Metal propellers should be maintained in accordance with the propeller manufacturer's instructions.

(b) PROPELLER HUBS.

(1) GENERAL. The hubs of propellers submitted for inspection and repair should be disassembled, and all parts should be cleaned in accordance with the manufacturer's recommendations. An inspection of the parts should be made to determine that the critical dimensions are within the manufacturer's specified tolerances. Particular care should be taken to check the 90-degree relation between shaft bore and blade socket centerline and track of the blade sockets, as these are the dimensions which are most likely to be affected by accidents. Any hub which is sprung should be rejected. Worn or damaged parts should be replaced. Stress raisers such as cuts, nicks, or tool marks should be carefully stoned out or the part rejected. Welding is not permissible, and remachining is permissible only when covered by a manufacturer's service bulletin.

(2) MAGNETIC PARTICLE INSPECTION. After cleaning, steel hubs should be

minutely inspected for cracks by the wet or dry magnetic particle method at every major overhaul period regardless of the repair classification (minor or major repair). It is not necessary to remove the plating or special external finish for this inspection unless so specified in the manufacturer's recommended practice. A brief statement recording the inspection and its findings should be included in the aircraft logbook entry. A similar entry should be made in the repair records of the repair base. In the case of an air carrier, this is considered the equivalent of the logbook entry. Any crack is cause for rejection.

(3) FINISH OF HUB PARTS. Plated hub parts from which the plating has been removed should be replated after they have been satisfactorily inspected. This plating should be done in accordance with the manufacturer's recommended practice. The use of zinc chromate primer on the external surfaces followed by a coating of aluminum lacquer in lieu of cadmium plating is considered equally acceptable.

(4) INSPECTION OF SPLINES AND CONE SEATS.

(i) WEAR. Splines and cone seats should be carefully inspected for signs of wear. Splines should be checked with a single key no-go gage made to plus 0.002 inch of the base drawing dimensions for spline land width. If the gage enters more than 20 percent of the spline area, the hub should be rejected and the local agent notified.

(ii) DISCOLORATION, PITTING, AND CORROSION. Cones and cone seats may show discoloration, pitting, and corrosion. Generally, corrosion and discolored spots may be removed by light lapping. Pitting is not grounds for rejection if 75 percent of the bearing area is not affected and the pitted areas are well dispersed about the cone bearing area.

(c) PROPELLER PITCH CONTROL EQUIPMENT. Governors, feathering accessories, and synchronizing equipment used to control the operation of certificated propellers should be inspected, assembled, and tested, in accordance with the manufacturer's recommended practice. It is recommended that all necessary replacement parts be obtained from the propeller manufacturer, and that only those repairs which are covered by manufacturers' service bulletins be made.

(d) ASSEMBLY OF PROPELLERS. Assembly of the propeller hub and/or blades should be accomplished in accordance with the manufacturer's recommendations. Clevis pins, bolts, and nuts which show wear or distortion should be replaced using standard aircraft parts. Cotter pins and safety wire should never be used a second time.

(e) REPAIR OF METAL PROPELLER BLADES.

(1) GENERAL. Metal propellers should never be operated with sharp edged dents, cuts, scars, scratches, nicks, or pits anywhere on the surface of a blade unless the manufacturer's instructions specifically permit it. Propellers incorporating either aluminum alloy or steel blades should be disassembled periodically and carefully inspected in accordance with the manufacturer's instructions. A damaged metal propeller is one which has been bent, cracked, or seriously injured. Minor surface dents, scars, nicks, or pits which are removable by field personnel, or are permitted by the manufacturer, are not considered to constitute a damaged propeller.

(i) NUMBER OF REPAIRS PERMITTED ON BLADES. More than one injury is not sufficient cause alone for rejection of a blade. A reasonable number of repairs per blade may be made and not necessarily result in a dangerous condition, unless their location with respect to each other is such as to form a continuous line of repairs that would materially weaken the blade.

(ii) NONREPAIRABLE BLADES. Damaged blades with model numbers which are on the manufacturer's lists of blades that cannot be repaired should be rejected.

(2) HOLLOW AND SOLID STEEL PROPELLER BLADES. A steel blade developing a crack of any nature in service should be returned to the manufacturer for inspection.

(i) DAMAGED STEEL BLADES. Damaged steel propeller blades—see CAM 18.20-14(e)(1)—should be repaired only by the manufacturers or by certificated repair agencies holding the appropriate ratings.

(ii) MINOR INJURIES TO STEEL BLADES. Minor injuries—see CAM 18.20-14(e)(1)—may be treated by authorized field personnel provided the manufacturer's recommendations are followed.

(3) ALUMINUM ALLOY PROPELLER BLADES. A damaged aluminum alloy propeller blade, see CAM 18.20-14(e)(1), should be repaired only by the manufacturer or by certificated repair agencies holding appropriate ratings. Minor injuries may be repaired as provided in CAM 18.20-14(e)(3)(vi) and subsections thereof.

(i) INSPECTION OF ALUMINUM ALLOY BLADES. Suspected cracks and all repairs of minor injuries should be etched with a warm 20 percent caustic soda solution and cleaned with a warm 20 percent nitric acid solution, and then examined with a three-power magnifying glass. Suspected cracks or defects should be repeatedly etched until their nature is determined. A crack will appear as a distinct black line. To avoid removal of an excess amount of metal, local etching should be accomplished at intervals during the process of removing suspected cracks. Upon completion of the repair, all effects of the etching process should be removed with fine emery paper.

Blades should be inspected at overhaul for cracks and material defects by either etching or anodizing. Etching is accomplished by immersing as much of the blade as possible, employing the same solutions as for local etching, with a warm water rinse between the caustic bath and the acid bath, and also a warm water rinse following the acid bath. All effects of the etching should be removed by polishing. The caustic and acid solutions should be maintained at a temperature of from 160° F. to 180° F. Some blades incorporate parts made of steel and other materials, and the caustic soda and the nitric acid must not be allowed to come in contact with these parts. The use of the fluorescent-penetrant method is recommended for the inspection of the shanks of such blades, as a supplement to the caustic etch.

The chromic acid anodizing process is superior to caustic etching for the detection of cracks and flaws and should therefore be used, whenever it is available, for general inspection of blades for material defects and for final checking of repairs performed during overhaul. The blades should be immersed in the anodizing bath as far as possible, but all parts not made of aluminum alloy must either be kept out of the chromic acid bath or be separated from the blade by nonconductive wedges or hooks. The anodizing treatment should be followed by a rinse in clear, cold, running water for 3 to 5 minutes, and the blades should then be dried as quickly as possible, preferably with an air blast. The dried blades should stand for at least 15 minutes before examination. Flaws (cold shuts or inclusions) will appear as fine black lines. Cracks will appear as brown stains caused by chromic acid bleeding out onto the surface. The blades may be "sealed" for improved corrosion resistance by immersing them in hot water (180° F. to 212° F.) for ½ hour. In no case should the blades be treated with hot water before the examination for cracks, since heating expands any cracks and allows the chromic acid to be washed away. A transverse (chordwise) crack or flaw of any size is cause for rejection. An excessive number of longitudinal flaws is cause for rejection. Any unusual condition or appearance revealed by caustic etching or anodizing should be referred to the manufacturer.

(ii) SHORTENING OF BLADES DUE TO DEFECTS. When the removal or treatment of defects on the tip necessitates shortening a blade, each blade used with it should likewise be shortened. Such sets of blades should be kept together (see Figure 14-5 for acceptable method). The shortened blades should be marked to correspond with the manufacturer's system of model designation to indicate propeller diameter. If, in making the repair, it is necessary to reduce the propeller diameter below the minimum diameter limit shown on the pertinent airplane specification, the repair should be submitted to a representative of the Civil Aeronautics Administration for approval. In such cases it may be necessary to investigate the performance characteristics of the airplane with the reduced diameter propeller installed.

(iii) BENT BLADES. Bent blades may be repaired only by the manufacturer or certificated repair agencies holding the appropriate ratings. The extent of a bend in face alinement should be carefully checked by means of a protractor similar to the one illustrated in Figure 14-1. Only bends not exceeding 20 degrees at 0.15 inch blade thickness to 0 degrees at 1.1 inch blade thickness may be cold-straightened. Blades with bends in excess of this amount require heat treatment. After straightening, the affected portion of the blade should be thoroughly inspected in accordance with CAM 18.20-14 (e) (3) (i).

(iv) MINIMUM LIMITS OF WIDTH AND THICKNESS FOR REPAIRED BLADES. For repaired blades the permissible reductions in width and thickness from minimum original dimensions allowed by the blade drawing and blade manufacturing specification are shown in Figure 14-4 for locations on the blade from the shank to 90 percent of the blade radius. Beyond the 90-percent blade radius point, the blade width and thickness may be modified as required.

The following tolerances are those listed in the blade manufacturing specification and govern the width and thickness of new blades. These tolerances are to be used with the pertinent blade drawing to determine the minimum original blade dimensions to which the reductions of Figure 14-4 may be applied.

		Manufacturing tolerance (inch)
Basic diameter less than 10 feet 6 inches:		
Blade width	from shank to 24-inch station	± 3/64
	from 30-inch station to tip	± 1/32
Blade thickness		± 0.025
Basic diameter 10 feet 6 inches to less than 14 feet 0 inches:		
Blade width	from shank to 24-inch station	± 1/16
	from 30-inch station to tip	± 1/32
Blade thickness	from shank to 24-inch station	± 0.030
	from 30-inch station to tip	± 0.025
Basic diameter 14 feet 0 inches and over:		
Blade width	from shank to 30-inch station	± 3/32
	from 36-inch station to tip	± 1/16
Blade thickness	from shank to 30-inch station	± 0.040
	from 36-inch station to tip	± 0.035

(v) REPAIR OF BLADE SHANKS. The shanks (roots or hub ends) of adjustable pitch blades should be etched locally in accordance with CAM 18.20-14(e) (3) (i). The shanks of such blades must be within drawing tolerances. The shanks of all other types of blades should be repaired in accordance with the manufacturer's instructions.

(vi) TREATMENT OF MINOR SURFACE INJURIES. Dents, cuts, scars, scratches, nicks, leading edge pitting, etc., should be removed or otherwise treated as explained below, provided their removal or treatment does not materially affect the strength, weight, or performance of the blade. When the repair of such injuries reduces the width or thickness of the blade below the limits allowed by CAM 18.20-14 (e) (3) (iv), the blade should be rejected. It is not permissible to peen down the edges of any injury wherein the operation will lap metal over the injury. Repair of blades identified by the manufacturer as being cold-worked (shot-blasted or cold-rolled) should be accomplished in accordance with the manufacturer's instructions, which may require peening.

(*a*) TREATMENT OF MINOR INJURIES ON THRUST AND CAMBER FACES. On the thrust and camber faces of blades the metal around any dents, cuts, scars, scratches, nicks, longitudinal surface cracks, and pits should be removed to form shallow saucer shaped depressions as shown in Figure 14–2 (view C). Care should be exercised to remove the deepest point of the injury and also to remove any raised metal around the edges of the injury as shown in Figure 14–2 (view A). All such repairs of minor injuries should be locally etched as explained in section 14 (e)(3)(i) to insure that no crack resulting from the injury remains. Any repair, performed in accordance with this paragraph, which results in a finished depression more than 1/8 inch in depth at its deepest point, 3/8 inch in width over-all, and 1 inch in length over-all, should be submitted to a certificated repair station for approval at the earliest possible opportunity and before an appreciable number of flying hours are accumulated.

(*b*) TREATMENT OF MINOR INJURIES AND WEAR ON BLADE EDGES. The following discussion specifically deals with leading edges of blades since these portions are the ones most likely to be damaged. Nevertheless, trailing edges of blades may be treated in substantially the same manner. Nicks, scars, cuts, etc., occurring on the leading edge should be smoothly rounded out as shown in Figure 14–2 (view B). Such repairs should be followed by local etching according to CAM 18.20–14(e)(3)(i) in order to detect any cracks resulting from the injuries. Any repair which results in a finished depression more than 1/4 inch deep (chordwise of the blade) and 1 inch long should be submitted to a certificated repair station for approval before an appreciable number of flying hours are accumulated. Blades that have the leading edges pitted from normal wear in service may be reworked by removing sufficient material to eliminate the pitting. In this case, the metal should be removed by starting well back from the edge, as shown in Figure 14–3, and working forward over the edge in such a way that the contour will remain substantially the same, avoiding abrupt changes in contour or blunt edges.

(vii) BALANCE. Upon completion of repairs, horizontal and vertical balance should be checked and any unbalance should be corrected as recommended by the manufacturer.

A coaxial hole is drilled in the butt end of certain aluminum alloy detachable blades for the application of lead to obtain static horizontal balance. The size of this hole should not be increased by the repair agency.

To effect vertical balance, only the manufacturer is permitted to drill and apply lead to an eccentric hole. The outside of this hole should be no closer than 1/4 inch to the nearest external blade surface.

As an alternate to drilling the two holes mentioned above, the manufacturer may have drilled a single eccentric hole having a diameter and depth conforming to the eccentric hole dimensions, given in the table below, for the application of lead. The outer edge of this hole should not be closer than 1 inch to the nearest external blade surface.

The ends of all balancing holes should be finished with a full sized drill having a spherical end to eliminate sharp corners. The sharp edges of all holes shuld be removed by a 1/32-inch chamfer.

The following table is included for inspectional information only as no drilling is to be done by anyone other than the manufacturer:

SIZE AND DEPTH OF BALANCING HOLES

Shank size	Maximum concentric hole diameter	Maximum concentric hole depth	Maximum eccentric hole depth (5/8 in. maximum diameter)
	Inch	*Inches*	*Inches*
00	7/16	2½	2¼
0–V2	19/32	3⅜	3
½	⅝	3⅝	3½
1	¾	4¼	4
1½	13/16	4⅞	4½
2	⅞	5½	5
3	31/32	6⅛	6

(viii) TRACK. The face alinement, or track, of the propeller should fall within the limits recommended by the manufacturer for new propellers.

(f) REPAIR OF WOOD PROPELLERS.

(1) GENERAL. Wood propellers should

be inspected for such defects as cracks, bruises, scars, warp, oversize holes in the hub, evidence of glue failure and separated laminations, sections broken off and defects in the finish. The tipping should be inspected for such defects as looseness or slipping, separation of soldered joints, loose screws, loose rivets, breaks, cracks, eroded sections, and corrosion.

(2) CAUSES FOR REJECTION. A wood propeller requiring repair should be carefully examined to insure that it can be restored to its original airworthy condition. Doubtful cases should be referred to the manufacturer and the proposed repair should be referred subsequently to a representative of the Civil Aeronautics Administration for approval. A propeller damaged to the following extent is considered unairworthy and should be scrapped immediately because repair is either impossible or uneconomical:

(i) A crack or deep cut across the grain of the wood.

(ii) Split blades or hubs.

(iii) Separated laminations. See CAM 18.20–14(f)(3)(ii).

(iv) An excessive number of screw or rivet holes.

(v) Oversize shaft hole.

(vi) An appreciable warp.

(vii) An appreciable portion of wood missing.

(viii) A crack, cut, or damage to the metal shank of adjustable-pitch wood or composition blades.

(3) METHODS OF MAKING REPAIRS TO WOOD BLADES.

(i) SMALL CRACKS PARALLEL TO THE GRAIN OF THE WOOD. Such cracks should be filled with glue thoroughly worked into all portions of the cracks, dried, and then sanded smooth and flush with the surface of the propeller. This also applies to small cuts.

(ii) SEPARATED LAMINATIONS. Whenever the glue joint of an outside lamination is open, the propeller may be repaired by removing the loose lamination and gluing on a new lamination of kiln dried wood of the same kind as the original lamination. It is not usually economical to attempt to repair separations between other laminations.

(iii) DENTS OR SCARS. Dents or scars which have rough surfaces or shapes that will hold a filler and will not induce failure may be filled with a mixture of glue and clean, fine, sawdust thoroughly worked and packed into the defect, dried and then sanded smooth and flush with the surface of the propeller. It is very important that all loose or foreign matter be removed from the place to be filled so that a good bond of the glue to the wood is obtained.

(iv) USE OF INLAYS. Inlays shown in Figure 14–6 of this manual may be used. Inlays should be of the same wood as the propeller blade; i. e., a yellow birch propeller should be inlaid with yellow birch not with white birch, and as near the same specific gravity as possible. Repair joints should conform with Figure 14–6 for taper of 10:1 from deepest point to feather edge or end of inlay. Measurements are taken along a straight line parallel to the grain or general slope of the surface on thrust and camber face. This rule applies also to the edge repairs. The grain of inlays should extend in the same direction as the grain of the propeller laminations.

Inlays should be made with a fishmouth, scarf, or butt joint. The permanency of the joint is in the order named, the fishmouth being preferable. Dovetail-type inlays should not be used. The number of inlays should not exceed one large, two medium, or four small widely separated inlays per blade. A trailing and a leading edge inlay should not overlap more than 25 percent, as shown in Figure 14–6.

On blades with normal sections from the mid-section to the tip, a cross-grain cut, up to 20 percent of the chord in length and one-eighth of the section thickness at the deepest point of damage may be repaired. On blades with thin sections, this depth should not exceed one-twentieth of the section thickness.

Narrow slivers up to 1/8 inch wide broken from the trailing edge at the wider portions of the blade may be repaired by sandpapering a new trailing edge, removing the least material possible, and fairing in a new trailing edge of smooth contour. Both blades should be narrowed by the same amount.

Near the hub or tip an inlay should be used and should not exceed, at its greatest depth, 5 percent of the chord.

In order to replace the wood worn away at the end of the metal tipping, enough of the metal should be removed to make the minimum repair taper 10:1 each way from the deepest point. Due to the convex leading edge of the average propeller, this taper usually works out 8:1. Repairs under the metal tipping should not exceed 7½ percent of the chord for butt or scarf joints, and 10 percent for fishmouth joints, with ¾-inch maximum depth for any repair.

(v) APPLICATION OF TIPPING FABRIC. When replacement of tipping fabric is necessary, it should be accomplished in the following manner:

(*a*) Launder the fabric (mercerized cotton airplane cloth) to remove all sizing.

(*b*) Cut a piece of fabric to approximate size required to cover both faces of outer portion of blade. The fabric should cover the same portion that the original fabric covered.

(*c*) Apply glue to the wood where the fabric is to be put on. Use a rather thick solution of the glue. Use resorcinol glue when the temperature of the workroom can be kept above 21° C. (70° F.); casein when the temperature cannot be kept above 21° C. (70° F.); and urea glue only when resorcinol is not available and the temperature is above 21° C. (70° F.).

(*d*) Put the fabric on glued surface, starting at the leading edge of the thrust face, and work toward the trailing edge. Fold the fabric around the trailing edge over the camber face, and toward the leading edge. Make a joint on the leading edge where it will be covered by the metal tipping. As the fabric is put on, smooth it out over the wood so as to prevent air bubbles or uneven glue underneath. Fabric must be perfectly flat on the blade.

(*e*) Trim excess fabric off with small scissors. Under no circumstances shall fabric be cut or scored with a knife.

(*f*) Allow the glue under the fabric to dry about 6 hours, then brush two coats of nitrate dope on the fabric, allowing ½ hour for drying time between coats. Allow another ½ hour for drying time and then sand the fabric lightly and brush a coat of pigmented dope over it. Lightly sand the uncovered portion of the wood and apply two coats of a good grade of moisture resistant spar varnish, allowing 12 to 16 hours drying time between coats.

(vi) METAL TIPPING. Tipping should be replaced when it cannot be properly repaired. Cracks in the narrow necks of metal between pairs of lobes of the tipping are to be expected and are not defects. All other cracks are defects that should be repaired, or eliminated by new tipping. Apply two coats of varnish to the wood to be covered by the metal tipping, if it has not been covered with fabric as above, and attach the tipping in the following manner.

(*a*) Obtain new tips and leading-edge strips, cut to size and formed to the approximate shape of the leading edge of the propeller. These pieces are usually supplied without holes so that the holes can be drilled in them to line up with the old screw and rivet holes in the propeller as below. If such material cut to shape is not available, the old tipping can be hammered out flat and used as a pattern to lay off a new tip. For this purpose use a piece of sheet metal of the same material and thickness as the old tip. Remove the burr from the cut edges of this piece.

(*b*) With a soft lead pencil, draw guide centerlines on the propeller extending about 4 inches from the centers of old screw and rivet holes. This procedure is followed to insure use of the original screw and rivet holes in the propeller. New holes should not be drilled except on replaced wood tips where there are no old holes or where the remaining portions of old holes may be plugged with glue. If new fabric has been applied to the propeller, puncture the fabric with a pointed tool at each hole and mark center guide lines on the fabric also. Coat screw and rivet holes in the propeller with varnish or with white lead and work the material into the holes.

(c) Lay the cut-out flat metal strip over the leading edge. Proceed to bend this metal down over the leading edge of the propeller, being careful that the metal extends an equal width on thrust and camber faces. This can be done by following the impressions of the old tipping lines. Numerous waves will occur in the metal, but these will be eliminated as the work progresses. Obtain several pieces of strong rubber tape, 4 feet long, ½ inch wide, and ¹⁄₁₆ inch thick.

While forming the metal, hold it in place on the propeller by wrapping the rubber tape around the blade. Start at the tip and work inboard, being careful not to cover the pencil lines placed on the propeller, which show the location of the rivet holes. While the metal is held in place, tap the leading edge with a rubber mallet, using moderate force to make sure the metal is seated against the wood along the nose of the leading edge. Smooth the metal by hammering it with a rubber mallet, backing up the opposite side of the blade with a laminated hardwood bucking block having an iron weight built in the center and a piece of leather fastened to the end on which the propeller bears. The block should measure about 2 by 4 inches. Start at the end of the blade and work toward the hub, moving the bucking block so that it is always immediately under the section being hammered. Continue to do this until the metal is well shaped to the profile of the propeller. Check to see that the metal has not moved from its original position. If this has happened, remove the rubber tape, reset the metal, and rewrap the rubber tape, thus forming the metal to the leading edge.

(d) With a centerpunch and a hammer, proceed to locate the old screw and rivet holes, using the pencil marks on the blade as a guide. Punch the metal approximately ¼ inch from the edge. After all holes have been located, remove the metal from the propeller. Drill screw and rivet holes in the metal with a ⅛-inch drill. File off burrs on the inside of the metal. Run the drill through the original rivet holes in the propeller, except those not to be used as above, in order to clean them out. Cut or saw slots in the metal at the original positions. (Refer to old tipping metal for locations of the slots.)

Place the metal leading edges on the blades they were formed to fit, and hold them in place with rubber tape. With a centerpunch as large as or slightly larger than the diameter of the screw and rivet heads, proceed to punch metal into the original countersunk holes in the wood so that the screw and rivet heads may be entered to the correct depth (not more than ¹⁄₃₂ inch below the surface of the metal). Use screws and rivets of the appropriate material. The screws should be one size larger than were originally in the propeller, and the rivets should be the solid flat, countersunk-head type. Insert screws and rivets in their respective holes. Install rivets with their heads on the thrust face of the propeller. After the rivets are tapped in place, cut off the excess length of the rivet, leaving ⅛ inch for heading. End cutters built up with solder to accurately measure this distance are very useful. While an assistant backs up the rivets with a steel bar 18 inches long and pointed to fit the rivet head, hammer the rivets either by hand or with a pneumatic hammer. Drive the screws, either by hand or with an electric screw driver.

Cut the metal of the cap-tip on the camber face of the propeller to the shape of the propeller tip. Bevel the edges by hand with a file. Trim off flat side of metal cap so that it extends about ³⁄₁₆ inch all around the tip of the propeller. Form a hardwood block to the shape of the thrust face of the propeller tip. Put metal tipping in place and clamp this block to the underside of the tip with a C-clamp. Turn this ³⁄₁₆ inch of metal up and over the camber face of the tip. Tighten and complete the lap joint. Mount the propeller blade solidly, with the thrust face up, on a stand supporting the blade at several points along its radius. With a hammer and a flat-faced tool, proceed to smooth the metal, starting at the nose of the leading edge and working toward the edge of the metal, until all wrinkles and high spots are removed. At the edge, use a caulking tool and, in the same manner, press the metal edge tightly against the wood. Turn the propeller over and repeat this operation on the camber face. Make sure that the thin tip is supported at all times when hammering. Apply solder over rivet and screwheads and over the metal seam of the tip of the propeller. Use 50–50 solder in wire form. Use muriatic acid as a flux when soldering brass. Use stainless steel soldering flux when soldering stainless steel tipping. File excess solder off and check the propeller balance while doing so. Polish the metal with a fine emery cloth or an abrasive drum driven by a flexible shaft.

(e) Vent the tipping by drilling three holes, No. 60 drill (.030), $\frac{3}{16}$ inch deep in the tip end. Vent holes should be parallel to the longitudinal axis of the blade.

(4) HUB, NECK, AND SHANK REPAIRS. The following subsections refer to propellers for the same engine horsepower and hub (see Figure 14–7).

(i) SMALL HUB DIAMETER, HEAVY NECKS AND SHANKS. Only the smallest of inlays should be used in the hub where there is any question of affecting the strength. The neck and shank are proportionately large in cross-section and fairly large repairs are possible, limited to a depth of about 5 percent of the section thickness.

(ii) SMALL TO MEDIUM DIAMETER HUB WITH EXCESSIVELY SMALL NECKS AND SHANKS. Only the smallest of inlays should be used in the hub where there is any question of affecting the strength. In the small shank area for propellers under 50 horsepower, cuts 5 percent deep may be filled with glue and sawdust. Over 50 horsepower, no inlay repairs should be made deeper than 2½ percent.

(iii) LARGE HUB DIAMETER, HEAVY NECKS AND SHANKS. Fairly large inlays are permissible on edges of hubs where cap laminations have crushed edges. Shank inlays should not exceed 7½ percent thickness of section for the heavy shanks, or 5 percent for the proportionately lighter shanks.

(iv) CRUSHED HUB FACES. Outside laminations, which have been crushed at the hub due to excessive drawing up of hub bolts, may be repaired by planing and sanding one hub face smooth, removing a lamination on the other hub face and replacing it with a new lamination, thus building the hub thickness up to the original thickness. It is permissible to replace both outer laminations if necessary and feasible.

(5) REPAIR OF ELONGATED BOLT HOLES IN PROPELLER HUB BOSS. It is permissible to repair elongated bolt holes by the insertion of a steel bushing around each bolt, as illustrated in Figure 14–8. In case method (A) or (B) is decided upon, the repairs should be made in accordance with the recommendations of the propeller metal hub manufacturer who is usually the engine manufacturer. In method (C), the bushing should be machined with an I.D. to fit the bolt snugly and an O.D. approximately ¼ inch larger than the bolt size. The bushing should be approximately ½ inch long. The face of the hub should be drilled with a hole concentric with the bolt hole and only to a sufficient depth to accommodate the bushing so that it does not protrude above the surface of the wood hub. The bushing should not be driven into the hub but should fit the hole in the hub with a clearance not exceeding .005 inch after moisture-proofing. The bushing hole should be protected from moisture by two coats of aluminum paint, varnish, glue, or other moisture-resistant coating.

(6) REPAIR OF ELONGATED BOLT HOLES IN PROPELLER HUB FLANGES. When the propeller bolt holes in a hub or crankshaft flange become damaged or oversize, it is permissible to make repairs by methods (A) or (B) in Figure 14–8, or by use of aircraft standard bolts $\frac{1}{16}$ inch larger than the original bolts.

(i) USE OF FLANGE BUSHINGS. Obtain from the engine or propeller hub manufacturer, suitable flange bushings with threaded or smooth bores, as illustrated in method (A) or (B) of Figure 14–8. Drill the flange and insert the bushings as recommended by the engine manufacturer. Drill the rear face of the propeller to accommodate the bushings, and protect the holes with *two coats of aluminum paint or other high* moisture-resistant coating. Use bolts of the same size as those originally used. Any of the following combinations may be used: Safety bolt and castellated nut, safety bolt *(drilled head) and threaded bushing, or un*drilled bolt and self-locking nut.

(ii) USE OF OVERSIZE BOLTS. Obtain suitable aircraft standard bolts 1/16 inch larger than the original bolts. Enlarge the *crankshaft propeller flange holes and the* propeller hub holes sufficiently to accommodate the new bolts without more than .005 inch clearance. Such reboring will be permitted *only once.* Further repairs of bolt holes may be made in accordance with methods (A) or (B) of Figure 14-8. (NOTE: Method (A) or (B) is preferred over the oversize bolt method, because a propeller hub flange redrilled in accordance with this latter method will always require the redrilling of all new propellers subsequently used with the redrilled flange.)

(7) FINISH. The finish, where necessary, should be renewed in accordance with the recommendations of the propeller manufacturer, or with a material which has satisfactory adhesion and high moisture-resistant properties.

(8) HORIZONTAL AND VERTICAL UNBALANCE. Final balance should be accomplished on a rigid knife-edge balancing stand in a room free from air currents. No persistent tendency to rotate from any position on the balance stand should be present.

(i) HORIZONTAL. Horizontal unbalance may be corrected by the application of finish or solder to the light blade. The light blade may be coated with a high grade of primer allowing for a finishing coat. After allowing each coat to dry 48 hours, the balance should be checked. Then, as may be necessary, either the required amount of finish should be removed by carefully sandpapering or an additional coat applied. The balance should be rechecked and sandpapered or additional finish applied as may be required to effect final balancing.

(ii) VERTICAL. Vertical unbalance may be corrected by applying putty to the *light side of the wood hub at a point on the* circumference approximately 90 degrees from the longitudinal centerline of the blades. The putty should be weighed and a brass plate weighing slightly more than the putty should be cut. The thickness of the plate should be from 1/16 to 1/8 inch depending on the final area, which must be sufficient for the required number of flathead attaching screws. The plate may be made to fit on the *hub face or to fit the shape of the light side* of the wood hub, and drilled and countersunk for the required number of screws. The plate should be attached and all of the screws tightened. After the plate is finally attached to the propeller, the screws should be secured to the plate by soldering the screw heads. The balance should be checked. All edges of the plate may be beveled to reduce its weight as necessary. The drilling of holes in the propeller and the insertion of lead or other material to assist in balancing will not be permitted.

(g) MODEL DESIGNATION. It is necessary to mark the name of the manufacturer and model designation on the repaired propeller in the event the original markings were removed during the repair or refinishing operations.

(h) IDENTIFICATION OF REPAIRING AGENCY. The Air Agency Certificate number, or name of agency making any repairs, especially on metal tipping, should be stamped or otherwise marked on the repaired propeller. It is recommended that a decalcomania giving both the repair agency's name and Air Agency Certificate number be used for this purpose.

18.20–15 WEIGHT AND BALANCE CONTROL. ***(CAA policies which apply to section 18.20.)***

(a) GENERAL. The purpose of the following information is to explain various items that are commonly involved in determining new weight and balance conditions. It indicates procedures by which the weight and balance of the airplane may be determined so that this factor may be controlled and properly used in the operation of the airplane within its approved limitations.

It is, of course, required that the airplane always be operated within the established limitations of performance, c.g. range, etc., as shown on the aircraft specification. It should be possible to determine these values in the event of a change in the airplane resulting from the removal or addition of equipment, or the making of repairs or alterations which affect the balance conditions of the airplane.

Information on which to base the record of weight and balance changes to the aircraft may be obtained from the Airplane Flight Manual, the Operations Limitations Forms ACA–309 or ACA–309a, or by actually weighing the airplane before undertaking an alteration or extensive repair. Footnotes 4, 5, and 6 and CAM 18.7–1(a) and 18.7–5(b) should be referred to for information on specific cases wherein weight and balance checks need not be made.

(1) TERMINOLOGY. The following terminology is used in the practical application of weight and balance control.

(i) MAXIMUM WEIGHT. The maximum weight is the maximum authorized weight of the aircraft and its contents as listed in the specifications.

(ii) EMPTY WEIGHT. The empty weight of an aircraft includes all operating equipment that has a fixed location and is actually in the aircraft. It includes the weight of the airframe, powerplant, required equipment, optional and special equipment, fixed ballast, full engine coolant, hydraulic fluid, and the fuel and oil as explained in CAM 18.20–15(b) (6) and (7). Additional information regarding fluids which may be contained in the aircraft systems and which must be included in the empty weight will be indicated in the pertinent aircraft specifications whenever deemed necessary.

(iii) USEFUL LOAD. The useful load is the empty weight subtracted from the maximum weight of the aircraft. This load consists of the pilot, crew if applicable, maximum oil, fuel, passengers, and baggage unless otherwise noted.

(iv) WEIGHT CHECK. A weight check consists of checking the sum of the weights of all items of useful load against the authorized useful load (maximum weight less empty weight) of the aircraft.

(v) DATUM. The datum is an imaginary vertical plane or line from which all horizontal measurements are taken for balance purposes with the aircraft in level flight attitude. The datum is indicated on most aircraft specifications. On some of the older aircraft where the datum is not indicated, any convenient datum may be selected. However, once the datum is located all moment arms must be taken with reference to it. Examples of typical locations of the datum are shown in Figure 15–1.

(vi) ARM (OR MOMENT ARM). The arm, or moment arm, is the horizontal distance in inches from the datum to the center of gravity of an item. The algebraic sign is plus (+) if measured aft of the datum and minus (—) if measured forward of the datum. Examples of plus and minus arms are shown in Figure 15–2.

(vii) MOMENT. Moment is the product of a weight multiplied by its arm. The moment of an item about the datum is obtained by multiplying the weight of the item by its horizontal distance from the datum. A typical moment calculation is given in Figure 15–3.

(viii) CENTER OF GRAVITY. The center of gravity is a point about which the nose-heavy and tail-heavy moments are exactly equal in magnitude. If the aircraft were suspended therefrom it would have no tendency to rotate in either direction (nose up or down). The weight of the aircraft (or any object) may be assumed to be concentrated at its center of gravity.

(ix) EMPTY WEIGHT CENTER OF GRAVITY. The empty weight c.g. is the center of gravity of an aircraft in its empty weight condition, and is an essential part of the weight and balance record. Formulas for determining the center of gravity for tail and nose-wheel type aircraft are given in Figure 15–4. Typical examples to determine the empty weight and c.g. for the tail-wheel and nose-wheel type aircraft are determined by computation in Figures 15–5 and 15–6.

(x) EMPTY WEIGHT CENTER OF GRAVITY RANGE. The empty weight center of gravity range is determined so that when the empty weight c.g. falls within this range the specification operating c.g. limits will not be exceeded under standard specification loading arrangements. In cases where it is possible to load an airplane in a manner not covered in the aircraft specification (i. e. extra tanks, extra seats, etc.), complete calculations should be accomplished, as outlined in CAM 18.20–15(c) (5) (i). The empty weight c.g. range, when applicable, is listed on the aircraft specifications.

(xi) OPERATING CENTER OF GRAVITY RANGE. The operating c.g. range is the distance between the forward and rearward center of gravity limits indicated on the pertinent aircraft specification. These limits were determined as the most forward and most rearward loaded c.g. positions at which the aircraft meets the requirements of the Civil Air Regulations. These limits are indicated on the specification in either percent of MAC or in inches from the datum. The c.g. of the loaded airplane must be within these limits at all times as illustrated in Figure 15–7.

(xii) MEAN AERODYNAMIC CHORD (MAC). The MAC is the mean aerodynamic chord of the wing. For weight and balance purposes it is used to locate the c.g. range of the aircraft. The location and dimensions of the MAC will be found in the Aircraft Specification, Flight Manual, or the Aircraft Weight and Balance Record on Form ACA–309 or ACA–309a.

(xiii) WEIGHING POINT. If the c.g. location is determined by weighing, it is necessary to obtain horizontal measurements between the points on the scales at which the airplane's weight is concentrated. If usual weighing practice is followed, a vertical line passing through the centerline of the axle will locate the point on the scale at which the weight is concentrated. This point is called the "Weighing Point". Other structural locations, capable of supporting the aircraft, such as jack pads on the main spar, may also be used. These points should be clearly indicated in the weight and balance record when used in lieu of the usual points. Typical locations of the weighing point are shown in Figure 15–8.

(xiv) MINIMUM FUEL. Minimum fuel for balance purposes is $\frac{1}{12}$ gallon per maximum-except-take-off-horsepower (METO), and is the minimum amount of fuel which should be used in weight and balance computations when low fuel might adversely affect the most critical balance conditions. To determine the weight of fuel in pounds divide the METO horsepower by 2.

(xv) FULL OIL. Full oil is the quantity of oil shown in the aircraft specifications as "oil capacity". Full oil should always be used as the quantity of oil when making the loaded weight and balance computations.

(xvi) TARE. Tare is the weight of chocks, blocks, stands, etc., used when weighing aircraft, and is included in the scale readings. Tare is deducted from the scale reading to obtain the actual aircraft weight.

(b) WEIGHING PROCEDURE. The following procedure should be followed when weighing an aircraft:

(1) The aircraft should be weighed inside a closed building to prevent error in scale reading due to wind.

(2) Excessive dirt, grease, moisture, etc., should be removed from the aircraft before weighing.

(3) If the center of gravity is to be determined, the aircraft should be placed in a level flight attitude.

(4) All items of equipment to be installed in the aircraft and included in the certificated empty weight should be in place for weighing. These items of equipment should be a part of the current weight and balance record. See CAM 18.20–15(c) (5) and 18.20–15(c) (5) (i).

(5) Scales should be properly calibrated, zeroed, and used in accordance with the scale manufacturer's instructions. The scales, and suitable supports for the aircraft if necessary, are usually placed under the wheels of a land plane, the keel of a seaplane float, or the skis of a ski plane. Other structural locations capable of supporting the air-

craft such as jack pads on the main spar also may be used. These points should be clearly indicated in the weight and balance data.

(6) Unless otherwise noted in the aircraft specification, the fuel system should be drained until the quantity indicator reads "zero" or "empty" with the aircraft in level flight attitude. The amount of fuel remaining in the tank, lines, and engine should be included in the empty weight. In special cases the aircraft may be weighed with full fuel in the fuel tanks provided a definite means of determining the exact weight of the fuel is available.

(7) Unless otherwise noted in the aircraft specification, the oil system should be completely drained with all drain cocks open. When weighed with full oil, actual empty weight equals the actual recorded weight less the weight of the oil in the oil tank (oil capacity in gallons × 7.5 pounds). All reports should indicate whether weights include full oil or oil drained (see Figure 15–9).

(8) Brakes should not be set while taking scale reading.

(9) Tare should be noted when the aircraft is removed from the scales.

(c) WEIGHT AND BALANCE COMPUTATIONS. It is often necessary, after completing an extensive repair or alteration, to establish by computation that the authorized weight or c.g. limits as shown on the aircraft specifications are not exceeded. The following information explains the significance of algebraic signs used in balance computations, outlines the loading conditions to check, and deals with equipment changes.

The aircraft specifications contain the following information relating to the subject:

- C.G. range.
- Empty weight c.g. range when applicable.
- MAC (Mean Aerodynamic Chord).
- Leveling means.
- Datum.
- Maximum weights.
- Number of seats and arm.
- Maximum baggage and arm.
- Fuel capacity and arm.
- Oil capacity and arm.
- Equipment items and arm.

(1) UNIT WEIGHTS FOR WEIGHT AND BALANCE PURPOSES.

Gasoline	6 pounds per U. S. gallon
Lubricating oil	7.5 pounds per U. S. gallon
Crew and passengers	170 pounds per person

(2) ALGEBRAIC SIGNS. Care should be exercised to insure retention of the proper algebraic sign (+ or —) throughout all balance computations and to always visualize the aircraft (for the sake of uniformity in these computations) with the nose to the left. In this position any arm to the left (forward) of the datum is minus and any arm to the right (rearward) of the datum is plus. Any item of weight added to the aircraft either side of the datum is a plus weight. Any weight item removed is a minus weight. When multiplying weights by arms, the answer is plus if the signs are alike and minus if the signs are unlike.

The following combinations are possible:

Items added forward of the datum—
(+) weight × (—) arm = (—) moment.

Items added to the rear of the datum—
(+) weight × (+) arm = (+) moment.

Items removed forward of the datum—
(—) weight × (—) arm = (+) moment.

Items removed rear of the datum—
(—) weight × (+) arm = (—) moment.

The total weight of the airplane is equal to the weight of the empty airplane plus the weight of the items added, minus the weight of the items removed.

The total moment is equal to the moment of the empty airplane combined with the individual moments of the items added or removed. In combining moments, plus moments are added and minus moments are subtracted.

(3) WEIGHT AND BALANCE EXTREME CONDITIONS. The weight and balance extreme conditions represent the

maximum forward and rearward c.g. positions for the aircraft. Information showing that the c.g. of the aircraft (usually in the fully loaded condition) falls between the extreme conditions should be included in the weight and balance data. The extreme conditions may be determined either by weighing or computation.

(i) FORWARD WEIGHT AND BALANCE CHECK. When a forward weight and balance check is made, it should be established that neither the maximum weight nor the forward c.g. limit listed in the aircraft specifications are exceeded. In making this check, the following information should be obtained:

(a) The weight, arm, and moment of the aircraft empty.

(b) The maximum weights, arms, and moments of the items of useful load which are located ahead of the forward c.g. limit.

(c) The minimum weights, arms, and moments of the items of useful load which are located ahead of the rearward c.g. limit.

A typical example of the computations necessary to make this check using the above data is shown in Figure 15-10.

(ii) REARWARD WEIGHT AND BALANCE CHECK. When a rearward weight and balance check is made, it should be established that neither the maximum weight nor the rearward c.g. limit listed in the aircraft specification are exceeded. In making this check, the following information should be obtained:

(a) The weight, arm, and moments of the aircraft empty.

(b) The maximum weights, arms, and moments of the items of useful load which are located aft of the rearward c.g. limit.

(c) The minimum weights, arms, and moments of the items of useful load which are located ahead of the rearward c.g. limit.

A typical example of the computation necessary to make this check using the above data is shown in Figure 15-11.

(4) LOADING CONDITIONS AND/OR PLACARDS. If the following items have not been covered in the weight and balance extreme condition checks, or are not covered by suitable placards in the aircraft, additional computations should be made. These computations should indicate the permissible distribution of fuel, passengers, and baggage which may be carried in the aircraft at any one time without exceeding either the maximum weight or the c.g. range. The conditions to check are:

(i) With full fuel, determine the number of passengers and baggage permissible.

(ii) With maximum passengers, determine the fuel and baggage permissible.

(iii) With maximum baggage, determine the fuel and the number and location of passengers permissible.

Examples of the computations for the above items are given in Figures 15-12, 15-13, and 15-14, respectively. The above cases are mainly applicable to the lighter-type personal aircraft. In the case of the larger-type transport aircraft, a variety of loading conditions is possible and it is usually necessary to have changes in the loading schedule approved separately by the CAA.

(5) EQUIPMENT LIST. A list of the equipment included in the certificated empty weight may be found in either the approved airplane operating manual or the Operations Limitations Form ACA-309a and its supplements. All required, optional, and special equipment installed in the aircraft at time of weighing and/or subsequent equipment changes should be entered in the weight and balance data.

Required equipment items are items so listed in the pertinent aircraft specification.

Optional equipment items are so listed in the pertinent aircraft items are so listed in be installed in the aircraft at the option of the owner.

Special equipment is any item not corresponding exactly to the descriptive information in the aircraft specification. This includes such items as flares, instruments, ash trays, radios, navigation lights, carpets, etc.

Required and optional equipment may be shown on the equipment list by making reference to the pertinent item number listed in the applicable specification only when they correspond exactly to that number item with reference to description, weight, and arm given in the specification. All special equipment items should be shown by making reference to the item by name, make, model, weight, and arm. When the arm for such an item is not available, it should be obtained by actual measurements.

(i) EQUIPMENT CHANGES. The owner should see that a continuous record for each aircraft is kept listing all changes affecting the weight, c.g. location, and equipment changes in order that a computed weight and c.g. location may be established at any time. An entry should be made on the equipment list indicating the items added, removed, or relocated, and the date accomplished. The identification of the repair agency should be included. Examples of items so affected are the installation of extra fuel tanks, seats, or baggage compartments. Figure 15–15 illustrates the effect on balance when equipment items are added within the acceptable c.g. limits and fore and aft of the established c.g. limits. Moment computations for typical equipment changes are given in Figure 15–16 and are also included in the sample weight and balance sheet in Figure 15–18.

(6) SAMPLE WEIGHT AND BALANCE REPORTS. Suggested methods of tabulating the various data and computations for determining the c.g., both in the empty weight condition and the fully loaded condition, are given in Figures 15–17 and 15–18, respectively. The data presented in Figure 15–17 have previously been computed in Figure 15–5 and represent a suggested means of recording this information. The data presented in Figure 15–18 have previously been computed in Figures 15–10 and 15–11 for the extreme loading conditions, and in Figure 15–16 for equipment change, and represent a suggested means of recording this information.

(d) LOADING SCHEDULE. The loading schedule should be kept with the aircraft and usually forms a part of the Form ACA–309 or the airplane flight manual. It includes instructions on the proper load distribution, such as filling of fuel and oil tanks, passenger seating, restrictions of passenger movement, distribution of cargo, etc.

Other means of determining safe loading conditions, such as the use of a graphical index, load adjuster, etc., are acceptable and may be used in lieu of the information in CAM 18.20–15(c) (4).

A separate loading condition should be computed when the aircraft is to be loaded in other than the specified conditions shown in the loading schedule.

FORMS

Forms to which reference has been made throughout Civil Aeronautics Manual 18 are reproduced in Appendix A.

APPENDIX A (Forms)

Form ACA-337 (11-48)

Form Approved.
Budget Bureau No. 41-R052.2.

DEPARTMENT OF COMMERCE
CIVIL AERONAUTICS ADMINISTRATION

REPAIR AND ALTERATION FORM (AIRCRAFT, PROPELLERS, ENGINES, INSTRUMENTS)

(SEE REVERSE SIDE OF THIS FORM FOR INSTRUCTIONS)

1. AIRCRAFT	MAKE	MODEL	SERIAL NO.	NATIONALITY AND REGISTRATION MARK
2. OWNER	NAME *(First, middle, last)*	ADDRESS *(Street and number, city, zone, and State)*		

3. FILL IN INFORMATION IN THIS ITEM ONLY FOR THE UNIT REPAIRED AND/OR ALTERED

UNIT	MAKE	MODEL	SERIAL NO.	NATURE OF WORK *(Check)*	
				MAJOR REPAIR	MAJOR ALTERATION
a. AIRCRAFT	(As described in item 1 above)				
b. PROPELLER BLADE OR HUB					
c. ENGINE					
d. INSTRUMENT	TYPE AND MANUFACTURER				

4. AIRCRAFT WEIGHT AND BALANCE DATA

This item must be completed by repair or alteration agency. However, in the case of a spare component, it will not be completed until such component is installed in an aircraft. At this time, it will be completed by the installing agency, if applicable.

**AFTER the repairs and/or alterations described below were made.*	EMPTY WEIGHT *(Pounds)**	EMPTY CENTER OF GRAVITY *(Inches from datum)**	USEFUL LOAD *(Pounds)**

5. KIND OF AGENCY WHICH MADE REPAIRS AND/OR ALTERATIONS *(Check)*

☐ MANUFACTURER ☐ APPROVED REPAIR STATION NO. ____________ *(Specify)* ☐ CERTIFIED MECHANIC

6. AGENCY	NAME	ADDRESS *(Street and number, city, zone, and State)*	DATE WORK ACCOMPLISHED

7. DESCRIPTION OF WORK (ALL WORK MUST BE ACCOMPLISHED IN ACCORDANCE WITH PART 18 OF THE CIVIL AIR REGULATIONS AND ITS ASSOCIATED CIVIL AERONAUTICS MANUAL 18.)

If more space is needed, continue on reverse, or attach separate sheets bearing aircraft registration mark.

☐ FORWARDED FOR ENGINEERING APPROVAL

I CERTIFY that the above statements are true and correct to the best of my knowledge.

(Signature of supervising mechanic) (Certificate number and rating) (Date)

TO BE COMPLETED BY CAA REPRESENTATIVES

☐ APPROVED	DESIGNEE'S SIGNATURE	NO.	DATE
☐ REJECTED	CAA AGENT SIGNATURE	☐ ACCEPTED ☐ REINSPECTED	DATE

(REVERSE OF FORM ACA–337)

INSTRUCTIONS

1. This form must be filled out in duplicate each time a major repair and/or alteration is made of an aircraft, propeller, engine, or instrument.
2. When repairs and/or alterations are made which affect the operation limitations set forth in the Airplane Flight Manual or Form ACA–309, the aircraft shall not be returned to service until the operation limitations have been corrected by an authorized representative of the CAA.
3. Certificated mechanics must, in all cases, obtain approval of the repair and/or alteration from the CAA representative prior to returning the article to service.
4. The manufacturer of an aircraft, engine, propeller, or instrument, and a certificated repair station holding the appropriate rating may return the article to service without prior approval of an authorized CAA representative, provided the alteration and/or repair does not change any of the operation limitations.
5. Repair agencies will be guided as follows when completing this form.

   a. For an Aircraft Repair and/or Alteration—Complete Items 1, 2, 3*a*, 4, 5, 6, and 7.

   Mechanic—Submit to CAA representative for inspection and approval prior to returning the article to service. Upon approval, the CAA representative will return the original copy to the mechanic who should submit it to the aircraft owner.

   Manufacturer or Approved Repair Station—Submit original to aircraft owner, forward copy to CAA district office or CAA agent prior to returning article to service.

   b. For a Component Installed in an Aircraft—Complete Items 1, 2, 3 (*b*, *c*, or *d*, whichever is applicable), 4, 5, 6, and 7. Distribute copies as in *a* above.

   c. For a Spare Component—Complete Items 3 (*b*, *c*, or *d*, whichever is applicable), 5, 6, and 7.

   Mechanic—Submit to CAA representative for inspection and approval. When approved, retain both copies of the form with the component until installation on an aircraft. At this time Items 1, 2, and 4 must be completed by the installing agency who will distribute the forms as follows: (No further approval of CAA is required, only a log-book entry by the installing agency is necessary.) After installation, original form should be submitted to aircraft owner, and copy forwarded to the nearest CAA district office or CAA agent.

   Manufacturer or Approved Repair Station—Handle same as for mechanics except that it is not necessary to submit to CAA representative for inspection or approval.

10—54010-2 U. S. GOVERNMENT PRINTING OFFICE

Form ACA-1226 (6-48) DEPARTMENT OF COMMERCE
CIVIL AERONAUTICS ADMINISTRATION

Budget Bureau No. 41-R495.5.
Approval expires Feb. 28, 1950.

MALFUNCTIONING AND DEFECTS REPORT
(OTHER THAN SCHEDULED AIR CARRIER)

INSTRUCTIONS: Fold, seal, and mail immediately. No postage required.

The Civil Aeronautics Administration requests the cooperation of all owners, pilots, operators, mechanics, inspectors, and investigators in reporting on this form difficulties experienced with aircraft structures, engines, propellers, and equipment, such as radio, instruments, fire extinguishers, brakes, instrument panel design, parachutes, improperly manufactured parts, etc.

Its purpose is to provide information which will enable the Civil Aeronautics Administration to take steps to prevent the recurrence of similar difficulties.

DO NOT SUBMIT THIS FORM IF a report of the same incident has been reported on Form CAB-453, Aircraft Accident Report.

1. DIFFICULTY OCCURRED	DATE	PLACE (*City and State*)

2. COMPLETE BOTH ITEMS IN THE FOLLOWING TABLE:

Item	NC No.	Make and Model	Serial No.	Hours Since Overhaul	Total Time (*Hours*)
a. Aircraft					
b. Engine					

3. SPECIFIC PART WHICH CAUSED DIFFICULTY. (Please make sketch showing manner of failure on other side of page.)

NAME OF PART	PART NO.	SERVICE TIME ON PART (*Hours*)	
		TOTAL	SINCE OVERHAUL

4. DESCRIBE IN DETAIL THE FAILED PART AND THE CIRCUMSTANCES UNDER WHICH THE DIFFICULTY OCCURRED.
(Attach photographs which clearly show the failure, or ship failed part if practicable, to assure appropriate corrective action; if shipped parts are large send these under separate cover properly identified as in items 1 and 2 above to: Civil Aeronautics Administration, Aircraft Service Analysis Staff, A-297, Washington 25, D. C.)

5. STATE PROBABLE CAUSE AND RECOMMENDATIONS TO PREVENT RECURRENCE. (In all cases include as much information as possible to indicate basis for analysis of cause.)

6. NAME (*Print*)	ADDRESS	DATE OF REPORT

CHECK WHICH: OWNER ☐ PILOT ☐ OPERATOR ☐ MECHANIC ☐ CAA DESIGNEE ☐ CAA INSPECTOR ☐ CAB INVESTIGATOR ☐

7. OWNER OF PLANE: PRIVATE OWNER ☐ FIXED BASE OPERATOR ☐ IRREGULAR CARRIER ☐

Fold and seal, with CAA address on the outside. Mail immediately. No postage required.

(REVERSE OF FORM ACA-1226)

FOLD HERE

DEPARTMENT OF COMMERCE
CIVIL AERONAUTICS ADMINISTRATION
WASHINGTON 25, D. C.

OFFICIAL BUSINESS

PENALTY FOR PRIVATE USE TO AVOID
PAYMENT OF POSTAGE, $300
(GPO)

CIVIL AERONAUTICS ADMINISTRATION

AIRCRAFT SERVICE ANALYSIS STAFF, A-297

WASHINGTON 25, D. C.

16—63347-2

FOLD HERE

AFTER FOLDING, MOISTEN GUMMED FLAP AND SEAL HERE

FOLD HERE

MAKE SKETCH SHOWING FAILURE

TABLES

Tables to which reference has been made throughout Civil Aeronautics Manual 18 are presented in Appendix B. These tables complement the various sections of CAM 18.20 and the numbers assigned to them have been developed and codified to each section of CAM 18.20. For instance, the first table relating to CAM 18.20–1 is numbered 1–1. This number (1–1) picks up the section number within CAM 18.20 and then by appending a dash followed by sequential numbering 1, 2, 3, etc., indicates which table within the section is being referenced. For example, Table 1–1 is the first table within section 1 of CAM 18.20.

TABLE 1-1—*Selection and Properties of Aircraft Wood*

Species of wood	Strength properties as compared to Spruce	Maximum permissible grain deviation (slope of grain)	Remarks
1	2	3	4
Spruce (Picea) Sitka (P. Sitchensis) Red (P. Rubra) White (P. Glauca).	100%	1:15	Excellent for all uses. Considered as standard for this table.
Douglas Fir (Pseudotsuga Taxifolia).	Exceeds spruce	1:15	May be used as substitute for spruce in same sizes or in slightly reduced sizes providing reductions are substantiated. Difficult to work with hand tools. Some tendency to split and splinter during fabrication and considerable more care in manufacture is necessary. Large solid pieces should be avoided due to inspection difficulties. Gluing satisfactory.
Noble Fir (Abies Nobiles)..	Slightly exceeds spruce except 8 percent deficient in shear.	1:15	Satisfactory characteristics with respect to workability, warping, and splitting. May be used as direct substitute for spruce in same sizes providing shear does not become critical. Hardness somewhat less than spruce. Gluing satisfactory.
Western Hemlock (Tsuga Heterophylla).	Slightly exceeds spruce	1:15	Less uniform in texture than spruce. May be used as direct substitute for spruce. Upland growth superior to lowland growth. Gluing satisfactory.
Pine, Northern White (Pinus Strobus).	Properties between 85 percent and 96 percent those of spruce.	1:15	Excellent working qualities and uniform in properties but somewhat low in hardness and shock-resisting capacity. Cannot be used as substitute for spruce without increase in sizes to compensate for lesser strength. Gluing satisfactory.
White Cedar, Port Oxford (Charaecyparis Lawsoniana).	Exceeds spruce	1:15	May be used as substitute for spruce in same sizes or in slightly reduced sizes providing reductions are substantiated. Easy to work with hand tools. Gluing difficult but satisfactory joints can be obtained if suitable precautions are taken.
Poplar, Yellow (Liriodendrow Tulipifera).	Slightly less than spruce except in compression (crushing) and shear.	1:15	Excellent working qualities. Should not be used as a direct substitute for spruce without carefully accounting for slightly reduced strength properties. Somewhat low in shock-resisting capacity. Gluing satisfactory.

(See notes on opposite page.)

NOTES (Table 1-1.)

1. Defects Permitted.
   - (a) *Cross grain.* Spiral grain, diagonal grain, or a combination of the two is acceptable providing the grain does not diverge from the longitudinal axis of the material more than specified in column 3. A check of all four faces of the board is necessary to determine the amount of divergence. The direction of free-flowing ink will frequently assist in determining grain direction.
   - (b) *Wavy, curly, and interlocked grain.* Acceptable if local irregularities do not exceed limitations specified for spiral and diagonal grain.
   - (c) *Hard knots.* Sound hard knots up to ⅜ inch in maximum diameter acceptable providing: (1) they are not in projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams or along the edges of flanges of box beams (except in lowly stressed portions); (2) they do not cause grain divergence at the edges of the board or in the flanges of a beam more than specified in column 3; and (3) they are in the center third of the beam and are not closer than 20 inches to another knot or other defect (pertains to ⅜-inch knots—smaller knots may be proportionately closer). Knots greater than ¼ inch should be used with caution.
   - (d) *Pin knot clusters.* Small clusters acceptable providing they produce only a small effect on grain direction.
   - (e) *Pitch pockets.* Acceptable in center portion of a beam providing they are at least 14 inches apart when they lie in the same growth ring and do not exceed 1½ inches length by ⅛ inch width by ⅛ inch depth and providing they are not along the projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams or along the edges of the flanges of box beams.
   - (f) *Mineral streaks.* Acceptable providing careful inspection fails to reveal any decay.

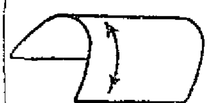
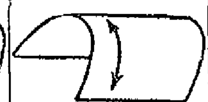
2. Defects Not Permitted.
   - (a) *Cross grain.* Not acceptable unless within limitations noted in 1(a).
   - (b) *Wavy, curly, and interlocked grain.* Not acceptable unless within limitations noted in 1(b).
   - (c) *Hard knots.* Not acceptable unless within limitations noted in 1(c).
   - (d) *Pin knot clusters.* Not acceptable if they produce large effect on grain direction.
   - (e) *Spike knots.* These are knots running completely through the depth of a beam perpendicular to the annual rings and appear most frequently in quarter-sawed lumber. Wood containing this defect should be rejected.
   - (f) *Pitch pockets.* Not acceptable unless within limitations noted in 1(e).
   - (g) *Mineral streaks.* Not acceptable if accompanied by decay (see 1(f)).
   - (h) *Checks, shakes, and splits.* Checks are longitudinal cracks extending, in general, across the annual rings. Shakes are longitudinal cracks usually between two annual rings. Splits are longitudinal cracks induced by artificially induced stress. Wood containing these defects should be rejected.
   - (i) *Compression wood.* This defect is very detrimental to strength and is difficult to readily recognize. It is characterized by high specific gravity, has the appearance of an excessive growth of summer wood, and in most species shows but little contrast in color between spring wood and summer wood. In doubtful cases the material should be rejected or samples should be subjected to a toughness machine test to establish the quality of the wood. All material containing compression wood should be rejected.
   - (j) *Compression failures.* This defect is caused from the wood being overstressed in compression due to natural forces during the growth of the tree, felling trees on rough or irregular ground or rough handling of logs or lumber. Compression failures are characterized by a buckling of the fibers that appear as streaks on the surface of the piece substantially at right angles to the grain, and vary from pronounced failures to very fine hair lines that require close inspection to detect. Wood containing obvious failures should be rejected. In doubtful cases the wood should be rejected or further inspection in the form of microscopic examination or toughness tests made, the latter means being the more reliable.
   - (k) *Decay.* All stains and discolorations should be examined carefully to determine whether or not they are harmless or preliminary or advanced decay. All pieces should be free from rot, dote, red heart, purple heart, and all other forms of decay.

TABLE 1-2—*Cold-Setting Synthetic-Resin Glues*[1]

Designation	Type	Approximate working life at 75° F. in hours
Amberlite PR-115	Resorcinol	4
Bakelite BC-17613	do	4
Bakelite BCU-1	Room-temperature urea	4
Bakelite BCU-5	do	4
Bakelite BCU-12772	do	4
Cascamite ANS	do	4
Cascamite 12	do	5-6
Cascamite 66	do	4
Cascamite 77	do	5-6
Cascamite 151	do	1¾
Cascophen RS-216	Resorcinol	3¾
Cascophen RS-224	do	3-4
Cascophen RS-232	do	3-4
Casco Resin No. 5	Room-temperature urea	4
Casco Resin No. 5	do	5-6
Casco Resin 135	do	8-9
Durez 12688	Resorcinol	3
Durite 2989	Room-temperature urea	5
Durite 3026	Resorcinol	3
Kaseno 2690	do	3
Lauxite PF90C	do	3
Lauxite 77-X	Room-temperature urea	4
Lauxite 81-MX	do	2-5
Lauxite 224	do	3
Le Page's Panite	Room-temperature urea	4
Penacolite G-1124	Resorcinol	2-3
Penacolite G-1131	do	2½-3
Penacolite G-1215	do	3
Perkins L-100	Room-temperature urea	4
Perkins D-110	do	5
Perkins DC-246	do	5
Perkins R-55	Resorcinol	2
Perkins RP-60	do	2
Phenac Resin Adhesive 703	do	5
Plaskon 201-2	Room-temperature urea	4
Plaskon 250-2	do	4
Plyophen 6000	Resorcinol	4
Synvaren PLS	do	2½-3½
Synvarite U	Room-temperature	4-5
Synvarol WR-513	do	2-3
Uformite 430	do	3-4
Uformite 500	do	3-4
Uformite CB-552	do	3-4
Urac 180	do	4-5
Urac Resin Adhesive 185	do	4½
USP Resorcinol	Resorcinol	3½-4
Weldwood	Room-temperature	3-4

[1] The inclusion of any glue in this table does not constitute an endorsement on the part of any government agency or assurance that it will meet applicable specifications.

TABLE 1-3—*Minimum Recommended Bend Radii for Aircraft Plywood*[1]

Plywood thickness		10 percent moisture content, bent on cold mandrels		Thoroughly soaked in hot water and bent on cold mandrels	
		At 90° to face grain 	At 0° or 45° to face grain 	At 90° to face grain 	At 0° or 45° to face grain 
(1)	(2)	(3)	(4)	(5)	(6)
Inch	No. plies	Inches	Inches	Inches	Inches
0.035	3	2.0	1.1	0.5	0.1
.070	3	5.2	3.2	1.5	.4
.100	3	8.6	5.2	2.6	.8
.125	3	12	7.1	3.8	1.2
.155	3	16	10	5.3	1.8
.185	3	20	13	7.1	2.6
.160	5	17	11	6	2
.190	5	21	14	7	3
.225	5	27	17	10	4
.250	5	31	20	12	5
.315	5	43	28	16	7
.375	5	54	36	21	10

[1] Columns (1) and (3) may also be used for determining the maximum thickness of single laminations for curved members.

TABLE 2–1—*Textile Fabric Used in Aircraft Covering*

Materials	Specification	Minimum Tensile Strength new (undoped)	Minimum Tearing Strength new (undoped)	Minimum Tensile Strength Deteriorated (undoped)	Thread Count Per Inch	Use and Remarks
Airplane cloth Mercerized cotton (Grade "A").	Society Auto. Engineers AMS 3806 (TSO-C15 references this Spec.).	80 lbs. per inch warp and fill	5 lbs. warp and fill	56 lbs. per inch	80 min. 84 max. warp and fill	For use on all aircraft. Required on aircraft with wing loadings greater than 9 psf. Placard never-exceed speeds greater than 160 mph.
.... do	AN–C–121	....do..	....do....	....do....	....do....	Alternate to AMS 3806.
Airplane cloth cellulose nitrate predoped.	AN–C–113	....do....	....do....	....do....	....do....	Alternate to AN–C–121 or AMS 3806 (undoped). Finish with cellulose nitrate dope.
Airplane cloth cellulose acetate butyrate, predoped.	AN–C–132	....do....	....do....	....do....	....do....	Alternate to AN–C–121 or AMS 3806 (undoped). Finish with cellulose acetate butyrate dope.
Airplane cloth mercerized cotton.	Society Auto. Engineers AMS 3804 (TSO-C14 references this Spec.).	65 lbs. per inch warp and fill	4 lbs. warp and fill	46 lbs. per inch	80 min. 94 max. warp and fill	For use on aircraft with wing loadings of less than 9 psf, and placarded never-exceed speeds of less than 160 mph.
Airplane cloth mercerized cotton.	Society Auto. Engineers AMS 3802 A	50 lbs. per inch warp and fill	3 lbs. warp and fill	35 lbs. per inch	110 max. warp and fill	For use on gliders with wing loading of 8 psf or less, and placard never-exceed speeds of 135 mph or less.
Glider fabric cotton.	A.A.F. No. 16128.	55 lbs. per inch warp and fill	4 lbs. warp and fill	39 lbs. per inch	80 min. warp and fill	Alternate to AMS 3802-A.

TABLE 2-2—*Miscellaneous Textile Materials*

Materials	Specification	Yarn Size	Minimum Tensile Strength	Yards Per lb.	Use and Remarks
Reinforcing tape, cotton.	AN-DDD-T-91a Type I.	....	150 lbs. per ½ inch width.	....	Used as reinforcing tape on fabric and under rib lacing cord. Strength of other widths approx. in proportion.
Lacing cord, prewaxed braided cotton.	AN-C-139	....	80 lbs. double.	310 min..	Lacing fabric to structures. Unless already waxed, must be lightly waxed before using.
Lacing cord, special cotton.	U. S. Army No. 6-27.	20/3/3/3	85 lbs. double.	....	.... do ....
Lacing cord, braided cotton.	AN-C-122	....	80 lbs. single.	170 min..	.... do ....
Lacing cord thread; linen and linen-hemp.	AN-T-47 Type II.	9 ply	59 lbs. single.	620 min..	.... do ....
		11 ply ...	70 lbs. single.	510 min..	.... do ....
Lacing cord thread; high-tenacity cotton.	AN-T-46. Style A. Type II.	Ticket No. 10.	62 lbs. single.	480 min..	.... do ....
Machine thread cotton.	Federal V-T-276b	20/4 ply..	5 lbs. single.	5000 normal.	Use for all machine sewing.
Hand sewing thread cotton.	V-T-276b. Type III B.	8/4 ply...	14 lbs. single.	1650 normal.	Use for all hand sewing. Use fully waxed thread.
Surface tape cotton (made from AN-C-121).	AN-T-48. Class I.	....	80 lbs./in.	....	Use over seams, leading edges trailing edges, outer edges and ribs, pinked, scalloped or straight edges.
Surface tape cotton.	Same as fabric used.	....	Same as fabric used.	....	Alternate to AN-T-48 Type I.

TABLE 3-1—*Oxyacetylene Welding Rod Properties*

Base material	Welding Rod Specification	Welding Rod Composition—%
Plain Carbon Steel and Low Alloy Steel Such As SAE 4130 (Not heat-treated after welding).	AAF No. 10286-D, Type II, Grade 1-G or Navy 46R4b, Class 1, Type C.	Carbon 0.06 Max. Manganese 0.25 Max. Phosphorus 0.04 Max. Sulphur 0.05 Max. Silicon[1] 0.08 Max.
Plain Carbon Steel and Alloy Steel (Heat-treated after welding).	AAF No. 10286-D, Type II, Grade 2-G or Navy 46R4b, Class 1, Type B.	Carbon 0.10-0.20 Manganese 1.00-1.20 Phosphorus 0.04 Max. Sulphur 0.05 Max. Silicon 0.20-0.30

[1] Silicon content not a requirement of Specification AAF No. 10286-D.

TABLE 3-2—*Arc Welding Electrode Properties*

Base material	Electrode Specification	Use
Low Carbon Steel and Alloy Steel (Not heat-treated after welding).	AN-E-9, Class A.	DC or AC Current, Shallow Penetration (Light Sections).
Low Carbon Steel and Alloy Steel (Not heat-treated after welding).	AN-E-9, Class B.	DC Reverse Polarity, Deep Penetration (Heavy Sections).
Alloy Steel (Heat-treated after welding).	AN-E-9, Class C.	DC or AC Current, Shallow Penetration (Light Sections).
Alloy Steel (Heat-treated after welding).	AN-E-9, Class D.	DC Reverse Polarity, Deep Penetration (Heavy Sections).

NOTE: Compositions not specified.

TABLE 3-3—*Recommended Bend Radii for 90° Bend—in Terms of Aluminum and Aluminum Alloy Sheet Thickness,* (t)

Alloy and temper	Approximate Thickness Inch					
	0.016 1/64	0.032 1/32	0.064 1/16	0.128 1/8	0.182 3/16	0.258 1/4
24S-O[1]	0	0	0	0	0-1t	0-1t
24S-T3[1,2]	1½t-3t	2t-4t	3t-5t	4t-6t	4t-6t	5t-7t
24S-T6[1]	2t-4t	3t-5t	3t-5t	4t-6t	5t-7t	6t-10t
52S-O	0	0	0	0	0-1t	0-1t
52S-H32	0	0	0	0-1t	0-1t	½t-1½t
52S-H34	0	0	0-1t	½t-1½t	1t-2t	1½t-3t
52S-H36	0-1t	½t-1½t	1t-2t	1½t-3t	2t-4t	2t-4t
52S-H38	½t-1½t	1t-2t	1½t-3t	2t-4t	3t-5t	4t-6t
61S-O	0	0	0	0	0-1t	0-1t
61S-T4	0-1t	0-1t	1½t-1½t	1t-2t	1½t-3t	2t-4t
61S-T6	0-1t	½t-1½t	1t-2t	1½t-3t	2t-4t	2t-4t
75S-O	0	0	0-1t	½t-1½t	1t-2t	1½t-3t
75S-T6[1]	2t-4t	3t-5t	3t-5t	4t-6t	5t-7t	6t-10t

[1] Alclad sheet can be bent over slightly smaller radii than the corresponding tempers of the uncoated alloy.
[2] Immediately after quenching, this alloy can be formed over appreciably smaller radii.

TABLE 3-4—*Number of Rivets Required for Splices (Single-lap Joint) in Bare 14S-T6, 24S-T3, 24S-T36, and 75S-T6 Sheet, Clad 14S-T6, 24S-T3, 24S-T36, and 75S-T6 Sheet, 24S-T4 and 75S-T6 Plate, Bar, Rod, Tube, and Extrusions, 14S-T6 Extrusions.*

Thickness "t" in Inches	No. of A17S-T3 Protruding Head Rivets Required Per Inch of Width "W"					No. of Bolts
	3/32	1/8	5/32	3/16	1/4	AN-3
0.016	6.5	4.9	....	....	....	....
.020	6.9	4.9	3.9	....	....	....
.025	8.6	4.9	3.9	....	....	....
.032	11.1	6.2	3.9	3.3	....	....
.036	12.5	7.0	4.5	3.3	2.4	....
.040	13.8	7.7	5.0	3.5	2.4	3.3
.051	....	9.8	6.4	4.5	2.5	3.3
.064	....	12.3	8.1	5.6	3.1	3.3
.081	....	....	10.2	7.1	3.9	3.3
.091	....	....	11.4	7.9	4.4	3.3
.102	....	....	12.8	8.9	4.9	3.4
.128	....	....	....	11.2	6.2	4.2

NOTES:

a. For stringers in the upper surface of a wing, or in a fuselage, 80 percent of the number of rivets shown in the table may be used.

b. For intermediate frames, 60 percent of the number shown may be used.

c. For single lap sheet joints, 75 percent of the number shown may be used.

ENGINEERING NOTES: *The above table was computed as follows:*

1. The load per inch of width of material was calculated by assuming a strip one inch wide in tension.
2. Number of rivets required was calculated for A17ST rivets, based on a rivet allowable shear stress equal to 40 percent of the sheet allowable tensile stress, and a sheet allowable bearing stress equal to 160 percent of the sheet allowable tensile stress, using nominal hole diameters for rivets.
3. Combinations of sheet thickness and rivet size above the heavy line are critical in (i. e., will fail by) bearing on the sheet; those below are critical in shearing of the rivets.
4. The number of AN-3 bolts required below the heavy line was calculated based on a sheet allowable tensile stress of 70,000 psi and a bolt allowable single shear load of 2,126 pounds.

TABLE 3-5—*Number of Rivets Required for Splices (Single-lap Joint) in 17ST, 17ST ALCLAD, 17SRT, and 17SRT ALCLAD Sheet, Plate, Bar, Rod, Tube, and Extrusions.*

Thickness "t" in Inches	No. of A17ST Protruding Head Rivets Required Per Inch of Width "W"					No. of Bolts
	3/32	1/8	5/32	3/16	1/4	AN-3
0.016	6.5	4.9	....	....	....	....
.020	6.5	4.9	3.9	....	....	....
.025	6.9	4.9	3.9	....	....	....
.032	8.9	4.9	3.9	3.3	....	....
.036	10.0	5.6	3.9	3.3	2.4	....
.040	11.1	6.2	4.0	3.3	2.4	....
.051	....	7.9	5.1	3.6	2.4	3.3
.064	....	9.9	6.5	4.5	2.5	3.3
.081	....	12.5	8.1	5.7	3.1	3.3
.091	....	....	9.1	6.3	3.5	3.3
.102	....	....	10.3	7.1	3.9	3.3
.128	....	....	12.9	8.9	4.9	3.3

NOTES:

a. For stringers in the upper surface of a wing, or in a fuselage, 80 percent of the number of rivets shown in the table may be used.

b. For intermediate frames, 60 percent of the number shown may be used.

c. For single lap sheet joints, 75 percent of the number shown may be used.

ENGINEERING NOTES: *The above table was computed as follows:*

1. The load per inch of width of material was calculated by assuming a strip one inch wide in tension.
2. Number of rivets required was calculated for A17ST rivets, based on a rivet allowable shear stress equal to 50 percent of the sheet allowable tensile stress, and a sheet allowable bearing stress equal to 160 percent of the sheet allowable tensile stress, using nominal hole diameters for rivets.
3. Combinations of sheet thickness and rivet size above the heavy line are critical in (i.e., will fail by) bearing on the sheet; those below are critical in shearing of the rivets.
4. The number of AN-3 bolts required below the heavy line was calculated based on a sheet allowable tensile stress of 55,000 psi and a bolt allowable single shear load of 2,126 pounds.

TABLE 3–6—*Number of Rivets Required for Splices (Single-lap Joint) in 52S (All Hardnesses) Sheet.*

Thickness "t" in Inches	No. of A17S-T3 Protruding Head Rivets Required Per Inch of Width "W"					No. of Bolts
	3/32	1/8	5/32	3/16	1/4	AN-3
0.016	6.3	4.7	...	...	...	...
.020	6.3	4.7	3.8	...	...	...
.025	6.3	4.7	3.8	...	...	...
.032	6.3	4.7	3.8	3.2	...	...
.036	7.1	4.7	3.8	3.2	2.4	...
.040	7.9	4.7	3.8	3.2	2.4	...
.051	10.1	5.6	3.8	3.2	2.4	...
.064	12.7	7.0	4.6	3.2	2.4	...
.081	...	8.9	5.8	4.0	2.4	3.2
.091	...	10.0	6.5	4.5	2.5	3.2
.102	...	11.2	7.3	5.1	2.8	3.2
.128	...	...	9.2	6.4	3.5	3.2

NOTES:

a. For stringers in the upper surface of a wing, or in a fuselage, 80 percent of the number of rivets shown in the table may be used.

b. For intermediate frames, 60 percent of the number shown may be used.

c. *For single lap sheet joints, 75 percent of the* number shown may be used.

ENGINEERING NOTES: *The above table was computed as follows*:

1. The load per inch of width of material was calculated by assuming a strip one inch wide in tension.
2. Number of rivets required was calculated for A17ST rivets, based on a rivet allowable shear stress equal to 70 percent of the sheet allowable tensile stress, and a sheet allowable bearing stress equal to 165 percent of the sheet *allowable tensile stress, using nominal hole* diameters for rivets.
3. Combinations of sheet thickness and rivet size above the heavy line are critical in (i. e., will fail by) bearing on the sheet; those below are critical in shearing of the rivets.

TABLE 4-1—*Aircraft Cable*

Cable size	Minimum breaking strength, pounds		
	Flexible 7 × 7 and	Flexible 7 × 19	Non-flexible 19 wire
	AN–C–43 carbon steel	AN–C–44 carbon steel	AN–C–76 carbon steel
1/32			185
1/16	480	480	500
5/64			800
3/32	920	920	1200
7/64			1600
1/8	2000	1900	2100
5/32	2800	2600	3300
3/16	4200	3900	4700
7/32	5600	5200	6300
1/4	7000	6600	8200
9/32	8000	8000	
5/16	9800	8200	12500
11/32	12500		
3/8	14400	12000	

TABLE 4-2—*Turnbuckle Safetying Guide*

Cable size	Type of wrap	Diameter of safety wire	Material (annealed condition)
1/16	Single	0.040	Copper, Brass, Galvanized or Tinned Steel, or Soft Iron
3/32	Single	.040	Copper, Brass, Galvanized or Tinned Steel, or Soft Iron
1/8	Single	.040	Stainless Steel
1/8	Double	.040	Copper, Brass, Galvanized or Tinned Steel, or Soft Iron
5/32 and greater	Double	.040	Galvanized or Tinned Steel, Soft Iron, or Stainless Steel
5/32 and greater	Double	.051	Copper, Brass

NOTES:

1. The swaged and unswaged turnbuckle assemblies are covered by AN Standard Drawings.
2. Certain of the AN Standard Swaged Terminal parts specify a safety wire hole size of .047 inch. This hole may be reamed sufficiently to accommodate the double .040 and .051 diameter wires.
3. In cases where the aircraft manufacturer recommends the use of a single wrap in lieu of a double wrap, the above recommendations concerning double wrap are considered inapplicable.

Table 5-1—*Recommended Nut Torques*

(CAUTION: These Torque Values Are Derived From Oil Free Cadmium-Plated Threads)

Tap size	Tension-type nuts AN-365 and AN-310	Sheer-type nuts AN-364 and AN-320	90,000 p.s.i. in bolts AN-365 and AN-310 nuts	(60% of column 4) AN-364 and AN-320 nuts
Fine-Thread Series				
8-36	12-15	7-9	20	12
10-32	20-25	12-15	40	25
¼-28	50-70	30-40	100	60
5/16-24	100-140	60-85	225	140
⅜-24	160-190	95-110	390	240
7/16-20	450-500	270-300	840	500
½-20	480-690	290-410	1100	660
9/16-18	800-1000	480-600	1600	960
⅝-18	1100-1300	600-780	2400	1400
¾-16	2300-2500	1300-1500	5000	3000
⅞-14	2500-3000	1500-1800	7000	4200
1-14	3700-5500	2200-3300[1]	10000	6000
1⅛-12	5000-7000	3000-4200[1]	15000	9000
1¼-12	9000-11000	5400-6600[1]	25000	15000
Coarse-Thread Series				
8-32	12-15	7-9	20	12
10-24	20-25	12-15	35	21
¼-20	40-50	25-30	75	45
5/16-18	80-90	48-55	160	100
⅜-16	160-185	95-100	275	170
7/16-14	235-255	140-155	475	280
½-13	400-480	240-290	880	520
9/16-12	500-700	300-420	1100	650
⅝-11	700-900	420-540	1500	900
¾-10	1150-1600	700-950	2500	1500
⅞-9	2200-3000	1300-1800	4600	2700

[1] Estimated corresponding values.

NOTE: The above torque loads may be used for all cadmium plated steel nuts of the fine or coarse thread series which have approximately equal number of threads and equal face bearing areas.

Table 7-1—*Numerical System for Steel Identification*

Type of Steel	*Numerals and Digits*
Carbon Steels	1xxx
Plain Carbon Steels	10xx
Free Cutting Steels	11xx
Manganese Steels (Manganese 1.60 to 1.90%)	13xx
Nickel Steels	2xxx
3.50% nickel	23xx
5.00% nickel	25xx
Nickel Chromium Steels	3xxx
9.70% nickel, 0.70% chromium	30xx
1.25% nickel, 0.60% chromium	31xx
1.75% nickel, 1.00% chromium	32xx
3.50% nickel, 1.50% chromium	33xx
Corrosion and Heat Resisting	30xxx
Molybdenum Steels	40xx
Chromium molybdenum steels	41xx
Nickel chromium molybdenum steels	43xx
Nickel molybdenum steels	
1.75% nickel, 0.25% molybdenum	46xx
3.50% nickel, 0.25% molybdenum	48xx
Chromium Steels	5xxx
Low chromium	51xx
Medium chromium	52xxx
Corrosion and heat resisting	51xxx
Chromium Vanadium Steels	6xxx
1.00% chromium	61xx
National Emergency Steels	8xxx
Silicon Manganese Steels	9xxx
2.00% silicon	92xx

TABLE 7-2—*Hardness Values for Steel*

Rockwell hardness		Vickers diamond pyramid hardness[1]	Brinell hardness[2]		Tensile strength, 1,000 lbs. per square inch
C–150 kg. load diamond	B–100 kg. load 1/16 ball		Tungsten carbide ball	Steel ball	
53	....	573	554	....	283
52	....	556	538	....	273
51	....	539	523	500	264
50	....	523	508	488	256
49	....	508	494	476	246
48	....	493	479	464	237
47	....	479	465	453	231
46	....	465	452	442	221
45	....	452	440	430	215
44	....	440	427	419	208
43	....	428	415	408	201
42	....	417	405	398	194
41	....	406	394	387	188
40	....	396	385	377	181
39	....	386	375	367	176
38	....	376	365	357	170
37	....	367	356	347	165
36	....	357	346	337	160
35	....	348	337	327	155
34	....	339	329	318	150
33	....	330	319	309	147
32	....	321	310	301	142
31	....	312	302	294	139
30	....	304	293	286	136
29	....	296	286	279	132
28	....	288	278	272	129
27	....	281	271	265	126
26	....	274	264	259	123
25	....	267	258	253	120
24	....	261	252	247	118
23	....	255	246	241	115
22	100.2	250	241	235	112
21	99.5	245	236	230	110
20	98.9	240	231	225	107
19[3]	98.1	235	226	220	104
18	97.5	231	222	215	103
17	96.9	227	218	210	102
16	96.2	223	214	206	100
15	95.5	219	210	201	99
14	94.9	215	206	197	97
13	94.1	211	202	193	95
12	93.4	207	199	190	93
11	92.6	203	195	186	91
10	91.8	199	191	183	90
9	91.2	196	187	180	89
8	90.3	192	184	177	88
7	89.7	189	180	174	87
6	89	186	177	171	85
5	88.3	183	174	168	84
4	87.5	179	171	165	83
3	87	177	169	162	82
2	86	173	165	160	81
1	85.5	171	163	158	80
0	84.5	167	159	154	78
....	83.2	162	153	150	76
....	82	157	148	145	74
....	80.5	153	144	140	72
....	79	149	140	136	70
....	77.5	143	134	131	68
....	76	139	130	127	66
....	74	135	126	122	64
....	72	129	120	117	62
....	70	125	116	113	60
....	68	120	111	108	58
....	66	116	107	104	56
....	64	112	104	100	54
....	61	108	100	96	52
....	58	104	95	92	50
....	55	99	91	87	48
....	51	95	86	83	46
....	47	91	83	79	44
....	44	88	80	76	42
....	39	84	76	72	40
....	35	80	72	68	38
....	30	76	67	64	36
....	24	72	64	60	34
....	20	69	61	57	32
....	11	65	57	53	30
....	0	62	54	50	28

[1] Vickers hardness values obtained with following loads:

	Kilograms
918 to 171, inclusive	50
167 to 95, inclusive	30
91 to 62, inclusive	10

[2] Brinell hardness values obtained with 3,000 kg. load except tungsten carbide ball values 159 to 86, inclusive, obtained with 1,500 kg. load and from 83 to 54, inclusive, with 500 kg. load.

[3] Rockwell C values below 20 are not recommended for correlation; however, these values are sufficiently accurate to indicate the trend of relationship.

TABLE 7-3—*Hardness Values for Aluminum Alloys*

Material commercial designation	Hardness temper	Brinell hardness No.
2S (pure aluminum)	O	23
	H	44
3S	O	28
	H	55
14S, forging	T	125
17S	O	45
	T	100
	RT	110
24S	O	42
	T	105
	RT	116
25S, forging	O	80
	W	100
	T	90
A51S, forging	T	90
52S	O	45
	H	85
61S	O	30
	T	95

TEMPER CODE

O — Annealed condition.
H — Maximum commercial degree of work hardening.
T — Fully heat-treated.
RT — Heat-treated and cold worked.
W — Quenched but not completely aged. W temper applies only to alloys requiring artificial aging to attain T condition.

TABLE 10-1—*Tube Data*

Tube O.D. (inches)	Wrench torque range for tightening tube nuts (inch pounds)		Minimum bend radii (inches)	
	Alum. alloy 2S½H, 52SO	Steel	Alum. alloy 2S½H, 52SO	Steel
⅛	....	....	⅜	....
3/16	....	30–70	7/16	21/32
¼	40–65	50–90	9/16	⅞
5/16	60–80	70–120	¾	1⅛
⅜	75–125	90–150	1 5/16	1 5/16
½	150–250	155–250	1¼	1¾
⅝	200–350	300–400	1½	2 3/16
¾	300–500	430–575	1¾	2⅝
1	500–700	550–750	3	3½
1¼	600–900	....	3¾	4⅜
1½	600–900	....	5	5¼
1¾	....	....	7	6⅛
2	....	....	8	7

TABLE 11-1—*Comparison of Copper and Aluminum Electric Cable Properties*

Cable size		Nominal conductor area (cir. mil)		Conductor resistance ohms/ft. at 20° C.		Current rating single conductor in free air
Cu	Al	Cu	Al	Cu	Al	Cu or Al
20	18	1119	1779	0.01025	0.010187	11
18	16	1779	2541	.00644	.007131	16
16	14	2409	4066	.00476	.004457	22
14	12	3830	6607	.00299	.002742	32
12	10	6088	10418	.00188	.001739	41
10	8	10443	16564	.00110	.001093	55
8	6	16864	28280	.00070	.000641	73
6	4	26813	42420	.000436	.000427	101
4	2	42613	67872	.000274	.000268	135
....	1	......	84840	....	.000214	155
2	1/0	66832	107464	.000179	.000169	181
1	2/0	81807	138168	.000146	.000133	211
1/0	3/0	104118	168872	.000114	.000109	245
2/0	4/0	133665	214928	.000090	.000085	283
3/0	....	167332	......	.000072	....	328
4/0	....	211954	......	.000057	....	380

APPENDIX C (Figures)

Figures to which reference has been made throughout Civil Aeronautics Manual 18 appear in Appendix C. The numbering system used in relation to tables, as explained in Appendix B, applies to the figures in this Appendix. For example, Figure 1–1 is the first figure within section 1 of CAM 18.20.

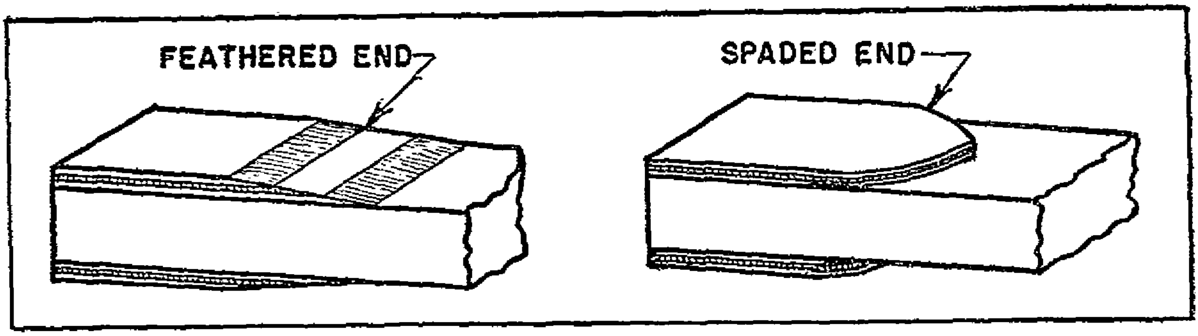


FIGURE 1–1. Tapering of Face Plates. Ref. CAM 18.20–1(a)(3)(ii)

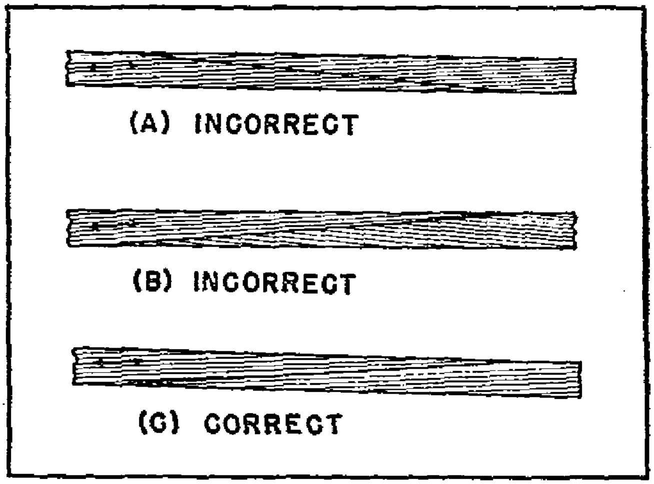


FIGURE 1–2. Consideration of Grain Direction When Making Scarf Joints. Ref. CAM 18.20–1(b)(2)

No grain deviation steeper than 1 in 15 should be present in an outer eighth of the depth of the spar. In an adjacent eighth, deviations involving steeper slopes, such as a wave in a few growth layers, are unlikely to be harmful. Local grain slope deviations in excess of those specified may be permitted in spar flanges only in the inner one-fourth of the flange depth.

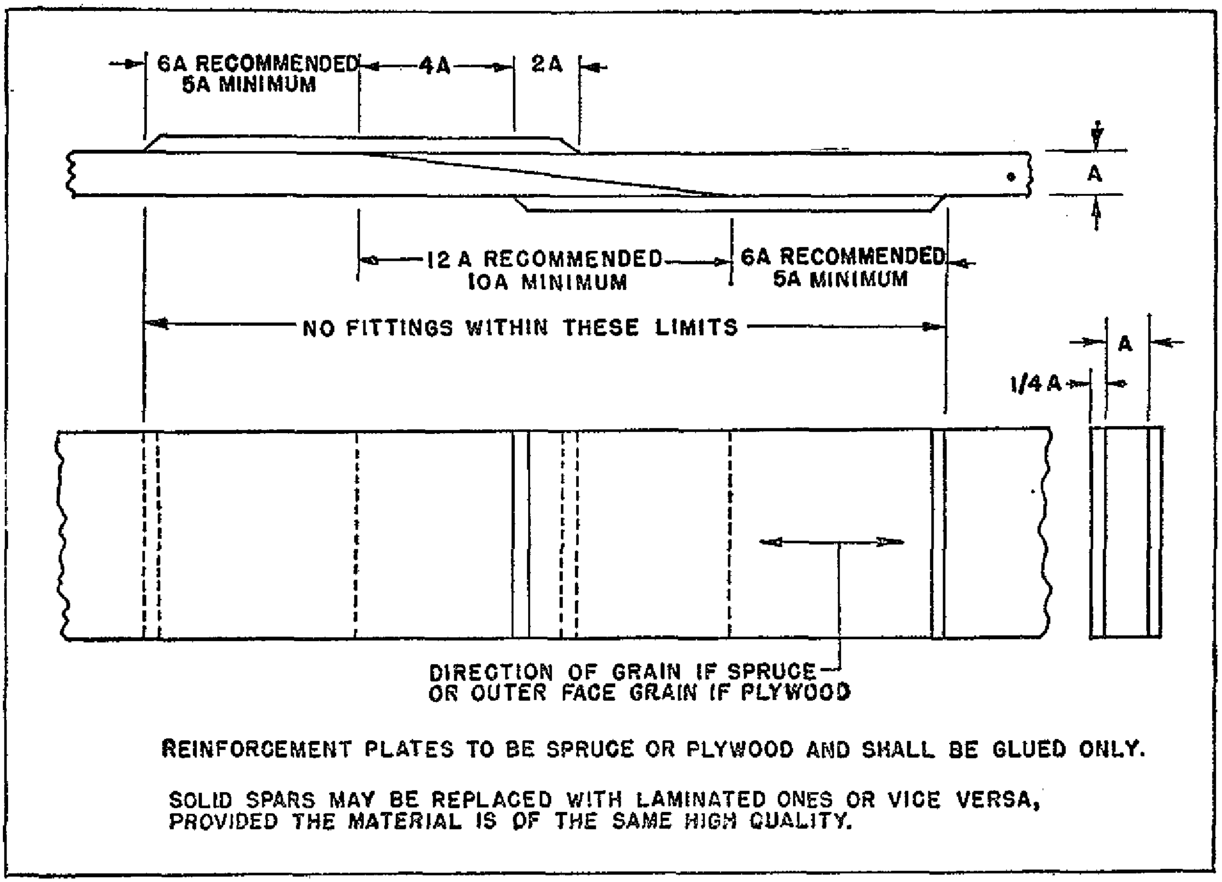


FIGURE 1–3. Method of Splicing Solid or Laminated Retangular Spars
Ref. CAM 18.20–1(c)(1)

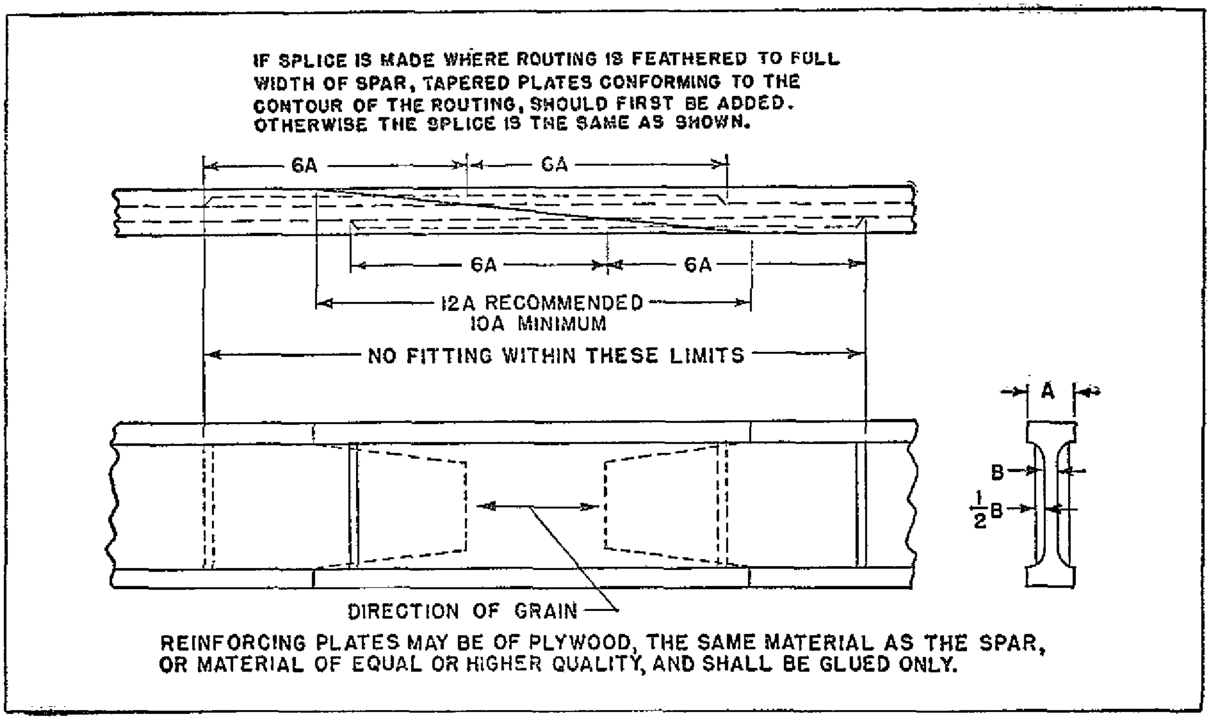


FIGURE 1–4. Method of splicing Solid "I" Spars. Ref. CAM 18.20–1(c)(1)

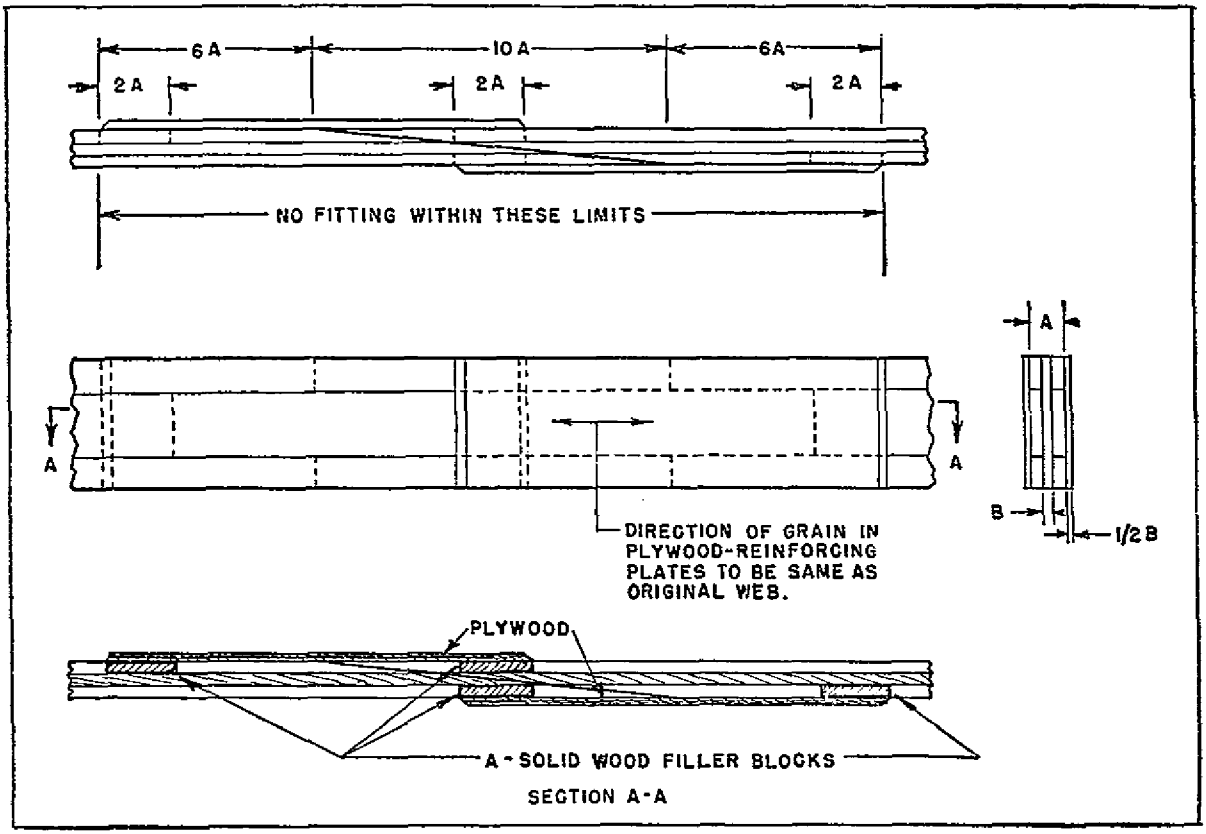


FIGURE 1–5. Repairs to Built-Up "I" Spar. Ref. CAM 18.20–1(c)(1)

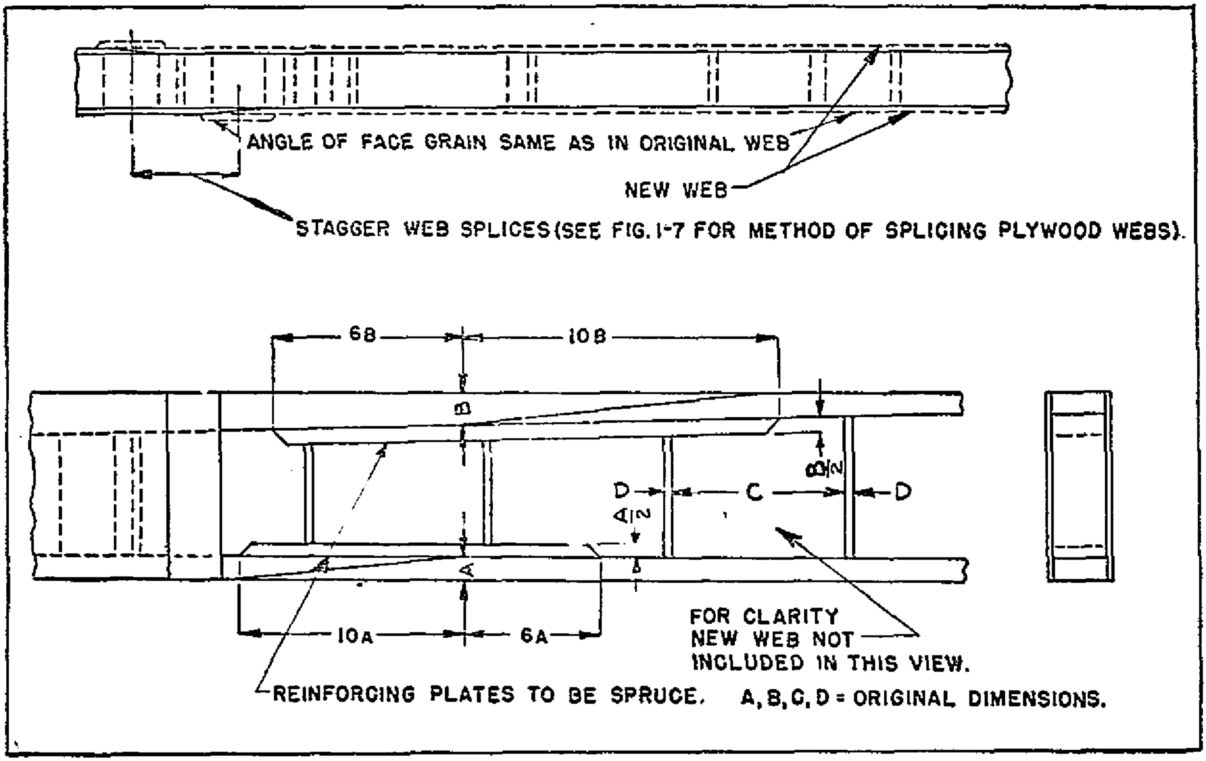
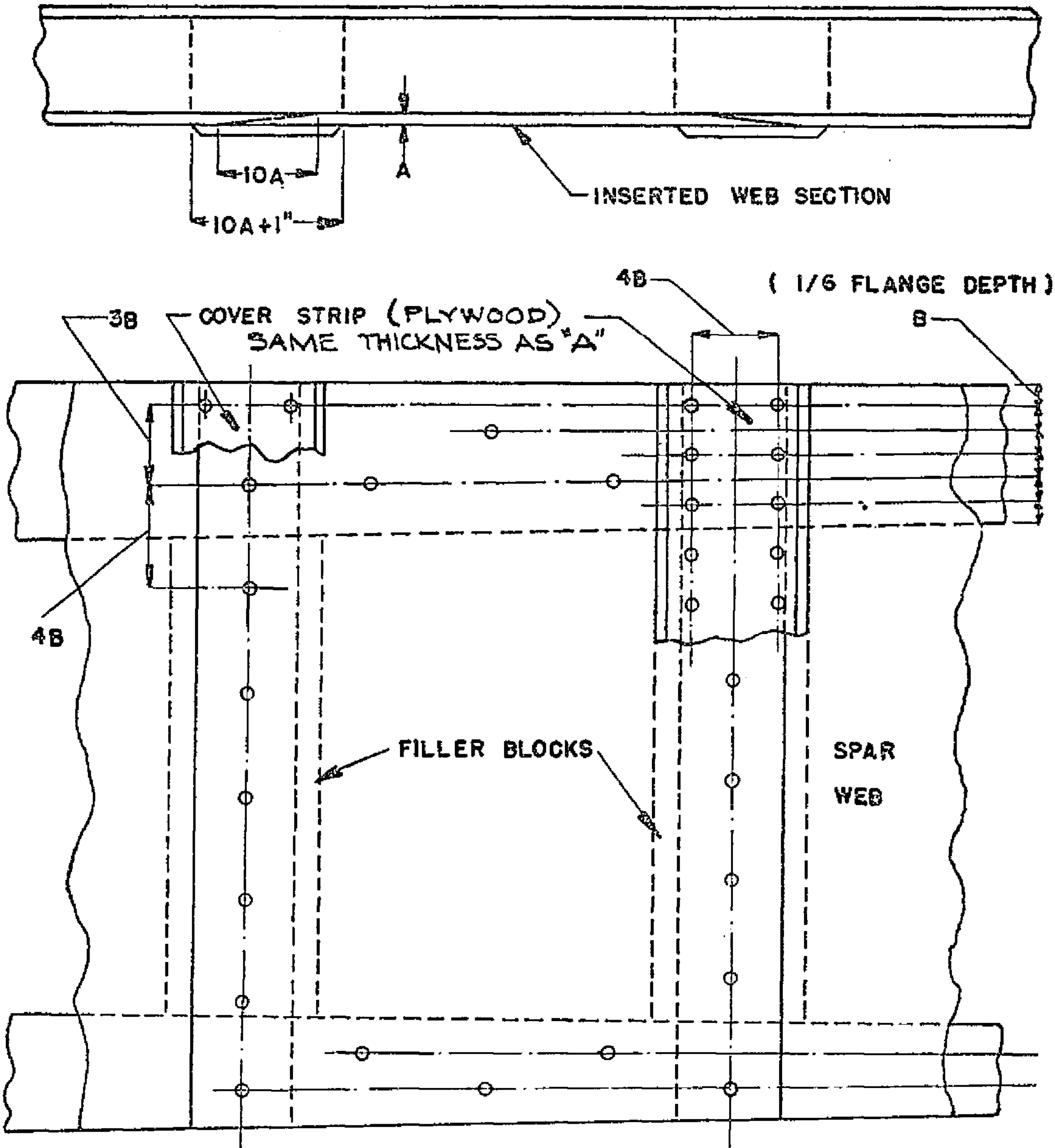


FIGURE 1–6. Method of Splicing Box-Spar Flanges (Plate Method). Ref. CAM 18.20-1(c)(1)



1. AFTER INSERTED WEB HAS BEEN GLUED AND NAILED IN PLACE, GLUE AND NAIL COVER STRIP OVER ENTIRE LENGTH OF SPLICE JOINTS.

2. SECTIONAL SHAPE OF FILLER BLOCKS MUST CONFORM EXACTLY TO TAPER OF SPAR. THEY MUST NOT BE TOO TIGHTLY FITTED OR WEDGING ACTION WILL LOOSEN EXISTING GLUE JOINTS OF WEBS TO FLANGES. IF TOO LOOSELY FITTED, CRUSHING OF WEB WILL OCCUR WHEN CLAMPING.

3. DIRECTION OF FACE GRAIN OF NEW PLYWOOD WEB AND COVER STRIPS TO BE SAME AS ORIGINAL WEB.

FIGURE 1-7. Method of Splicing Box-Spar Webs. Ref. CAM 18.20-1(c)(2)

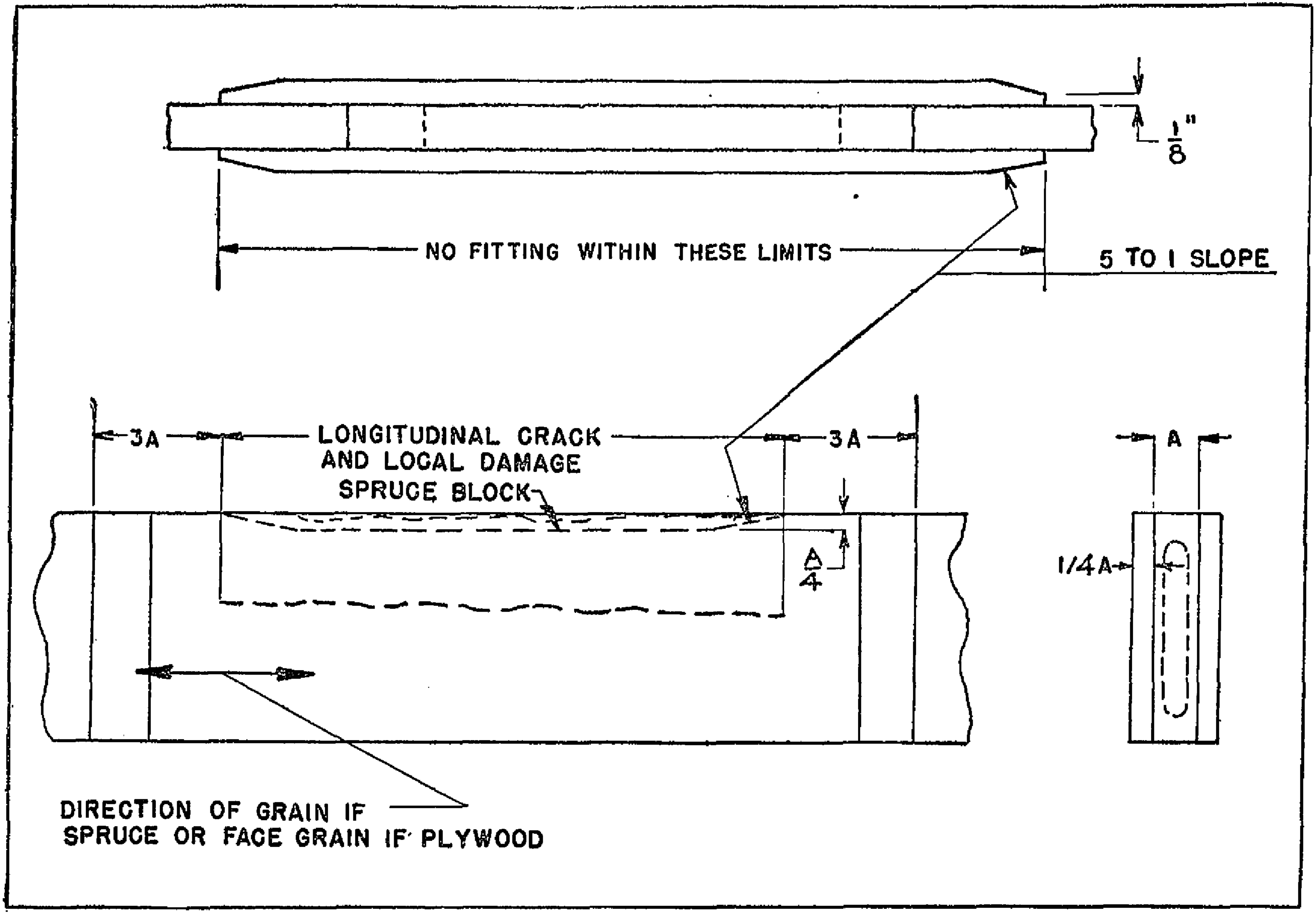


FIGURE 1-8. Method of Reinforcing a Longitudinal Crack and/or Local Damage in a Solid or Internally Routed Spar. Ref. CAM 18.20-1(c)(4)

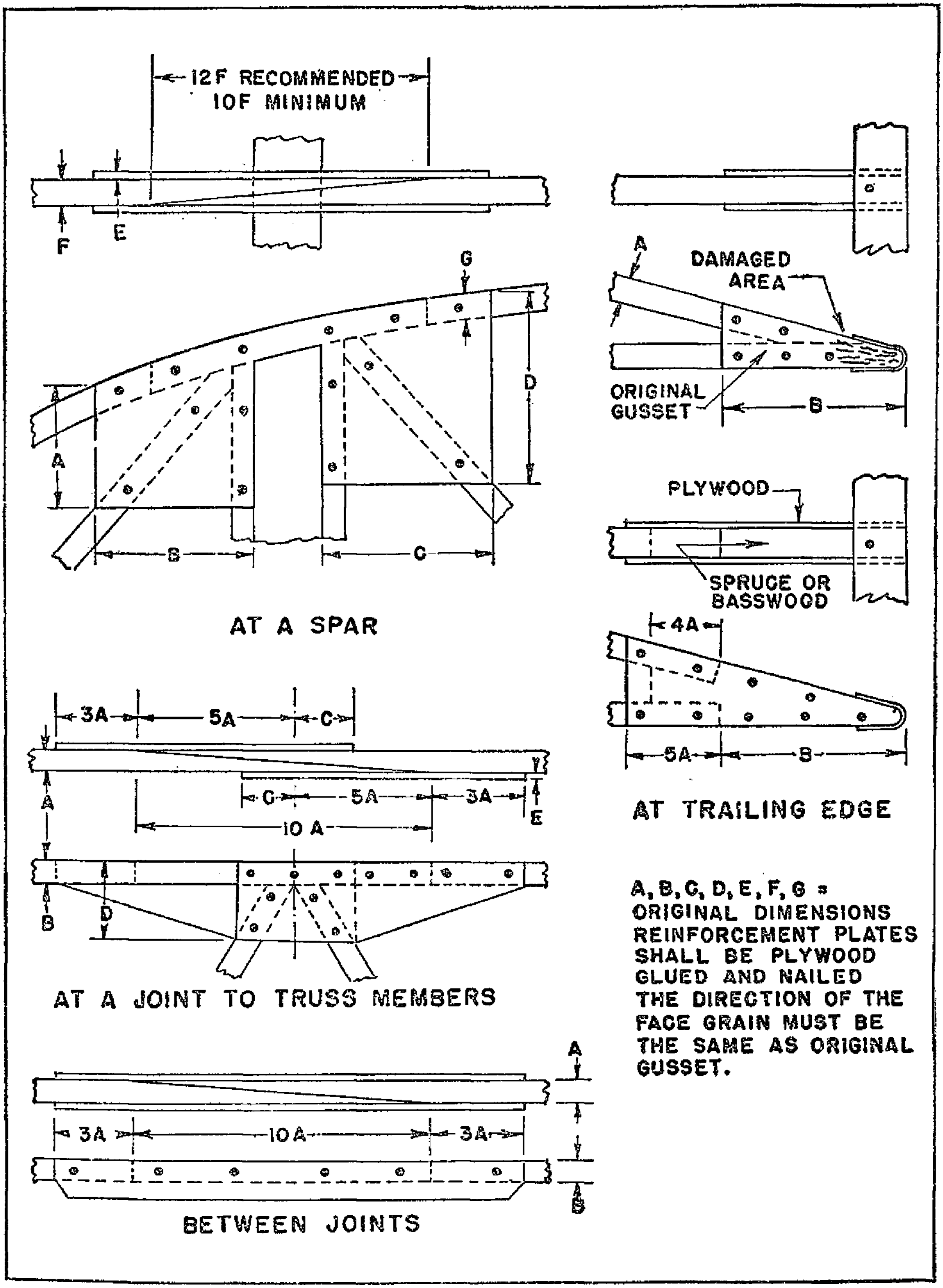


FIGURE 1-9. Repair of Wood Ribs. Ref. CAM 18.20-1(d)(2), 18.20-1(d)(3)

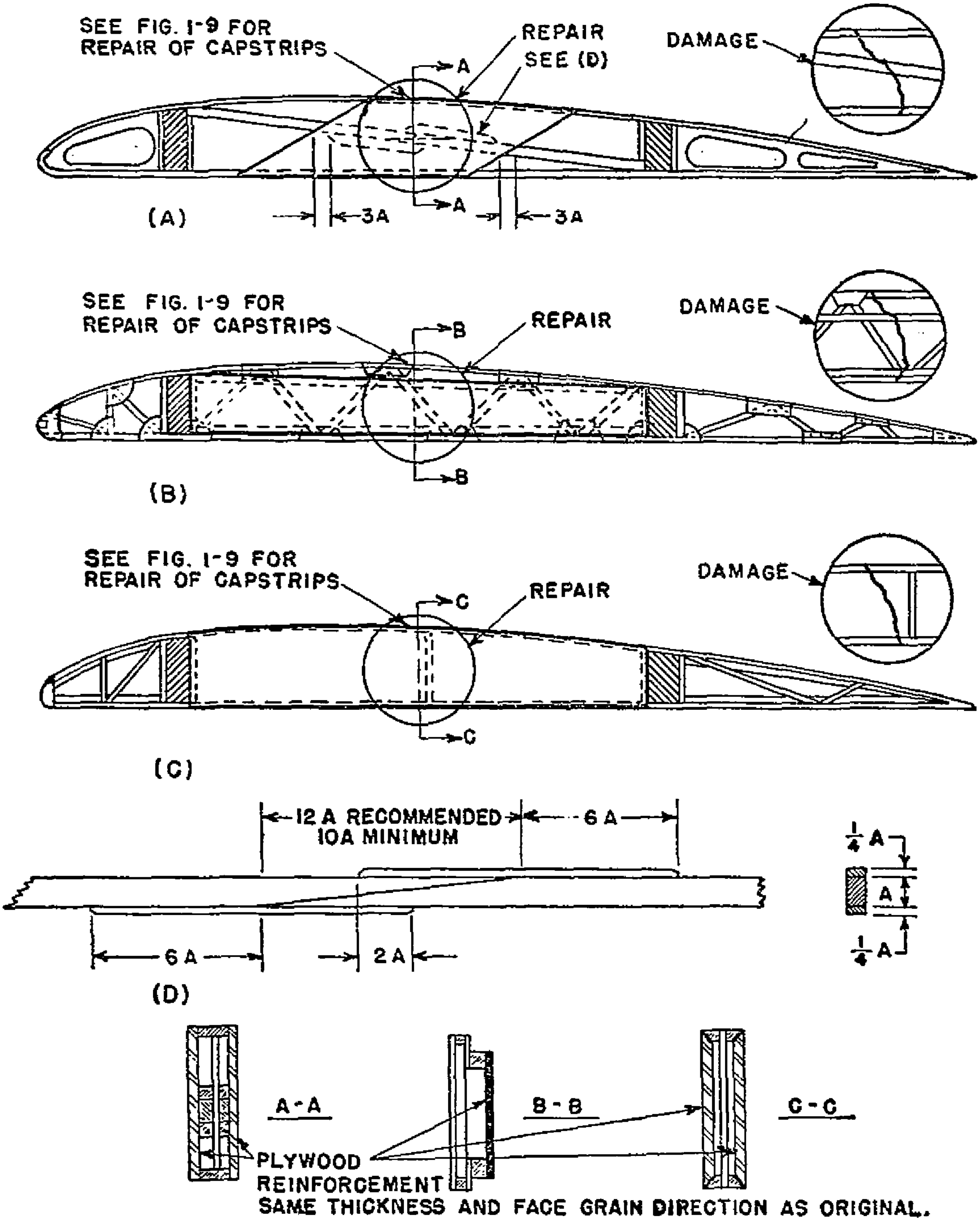


FIGURE 1-10. Typical Wing Compression Rib Repairs. Ref. CAM 18.20-1(d) (3)

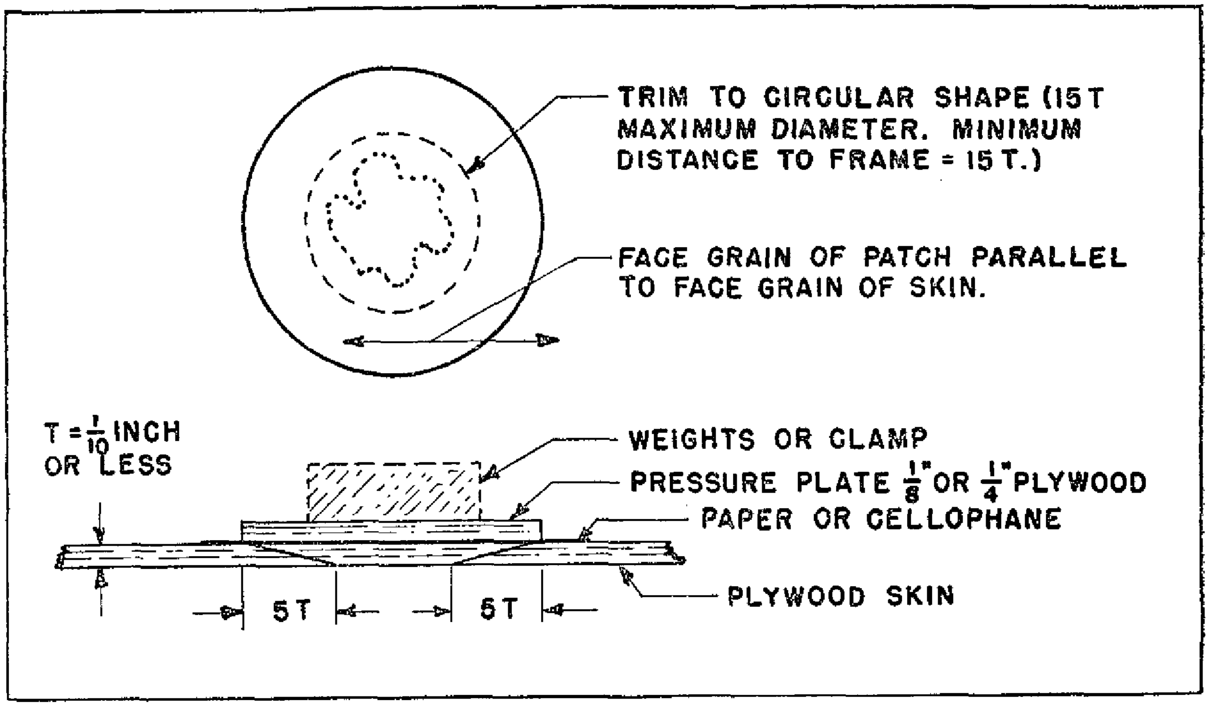
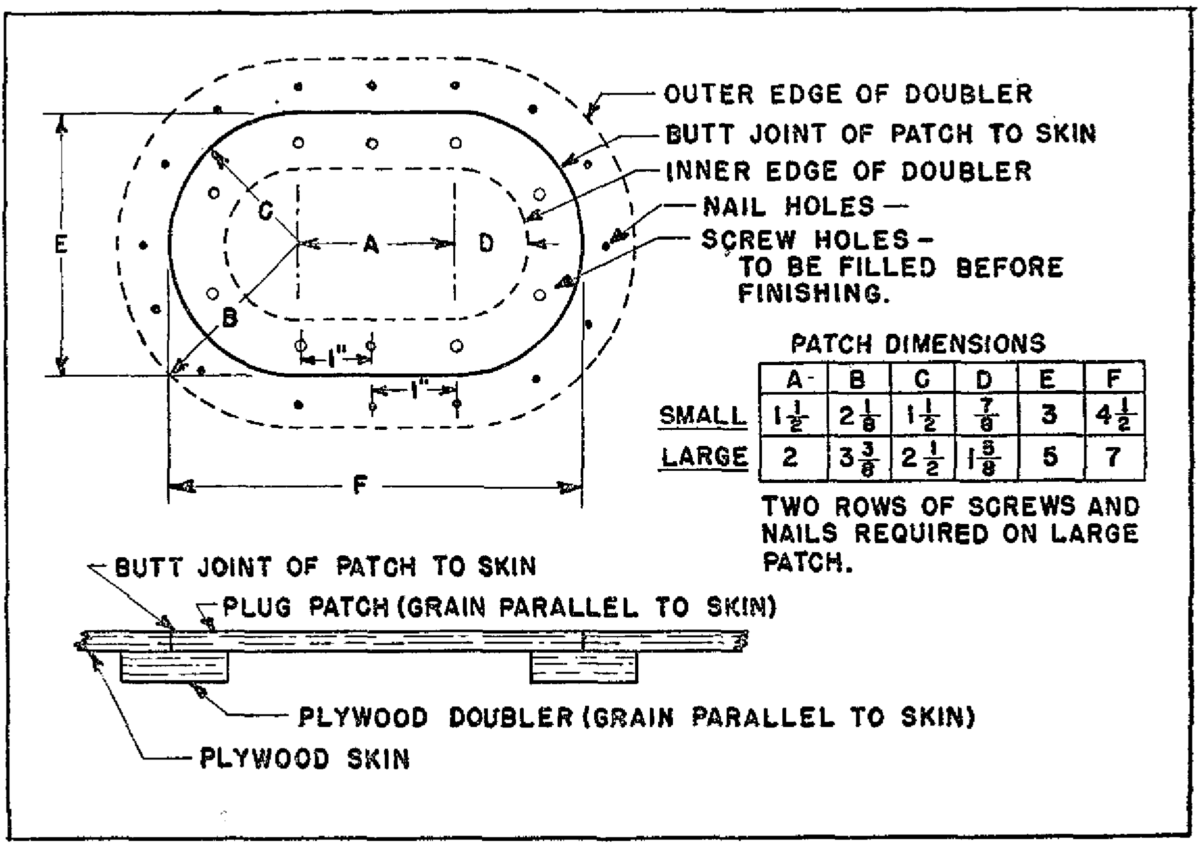


FIGURE 1-11. Splayed Patch. Ref. CAM 18.20-1(e)(2)



	A	B	C	D	E	F
SMALL	$1\frac{1}{2}$	$2\frac{1}{8}$	$1\frac{1}{2}$	$\frac{7}{8}$	3	$4\frac{1}{2}$
LARGE	2	$3\frac{3}{8}$	$2\frac{1}{2}$	$1\frac{5}{8}$	5	7

FIGURE 1-12. Oval Plug Patch Assembly. Ref. CAM 18.20-1(e)(5)(i), 18.20-1(e)(5)(ii)(a)

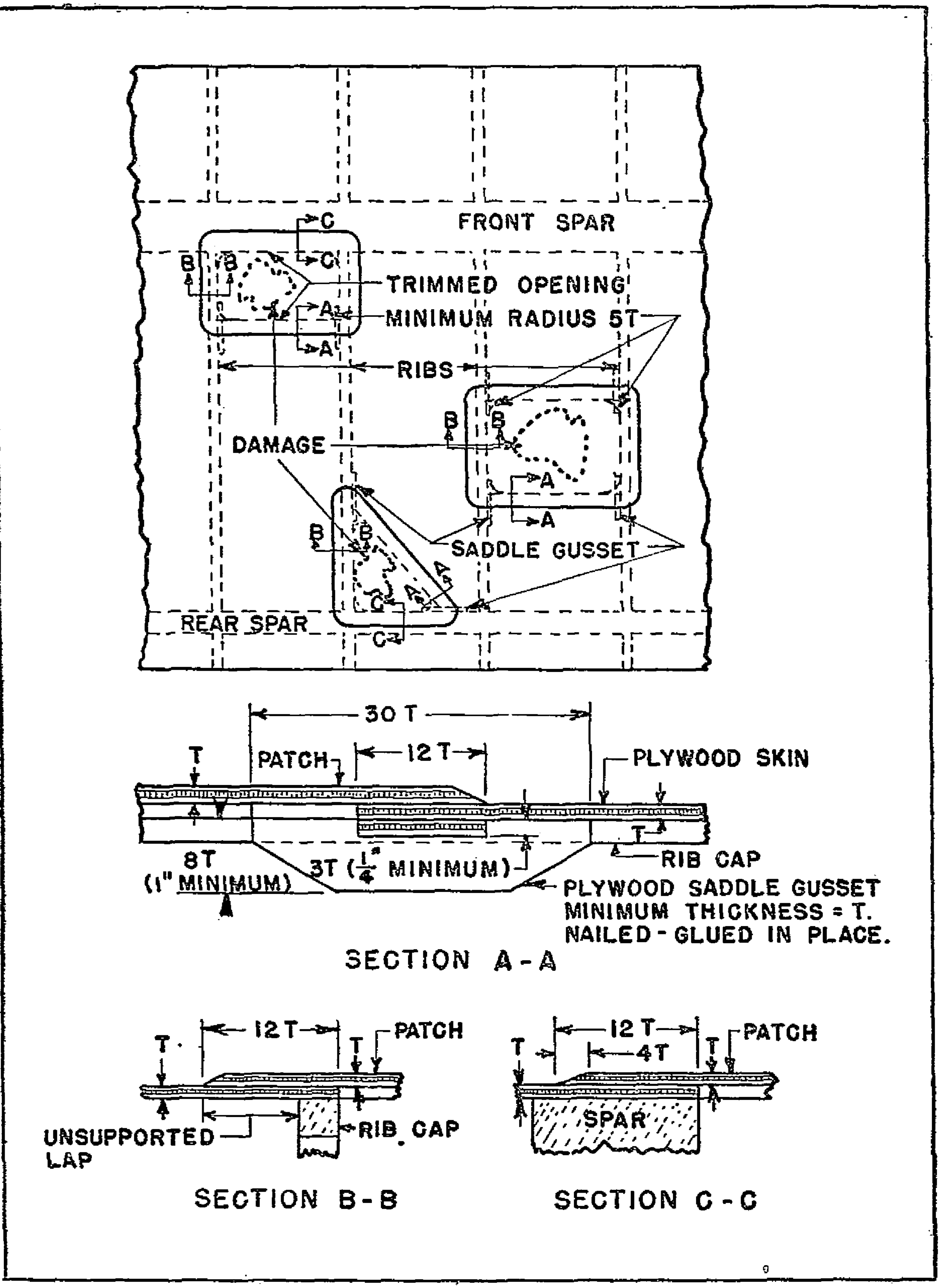


FIGURE 1-13. Surface Patches. Ref. CAM 18.20-1(e)(3)

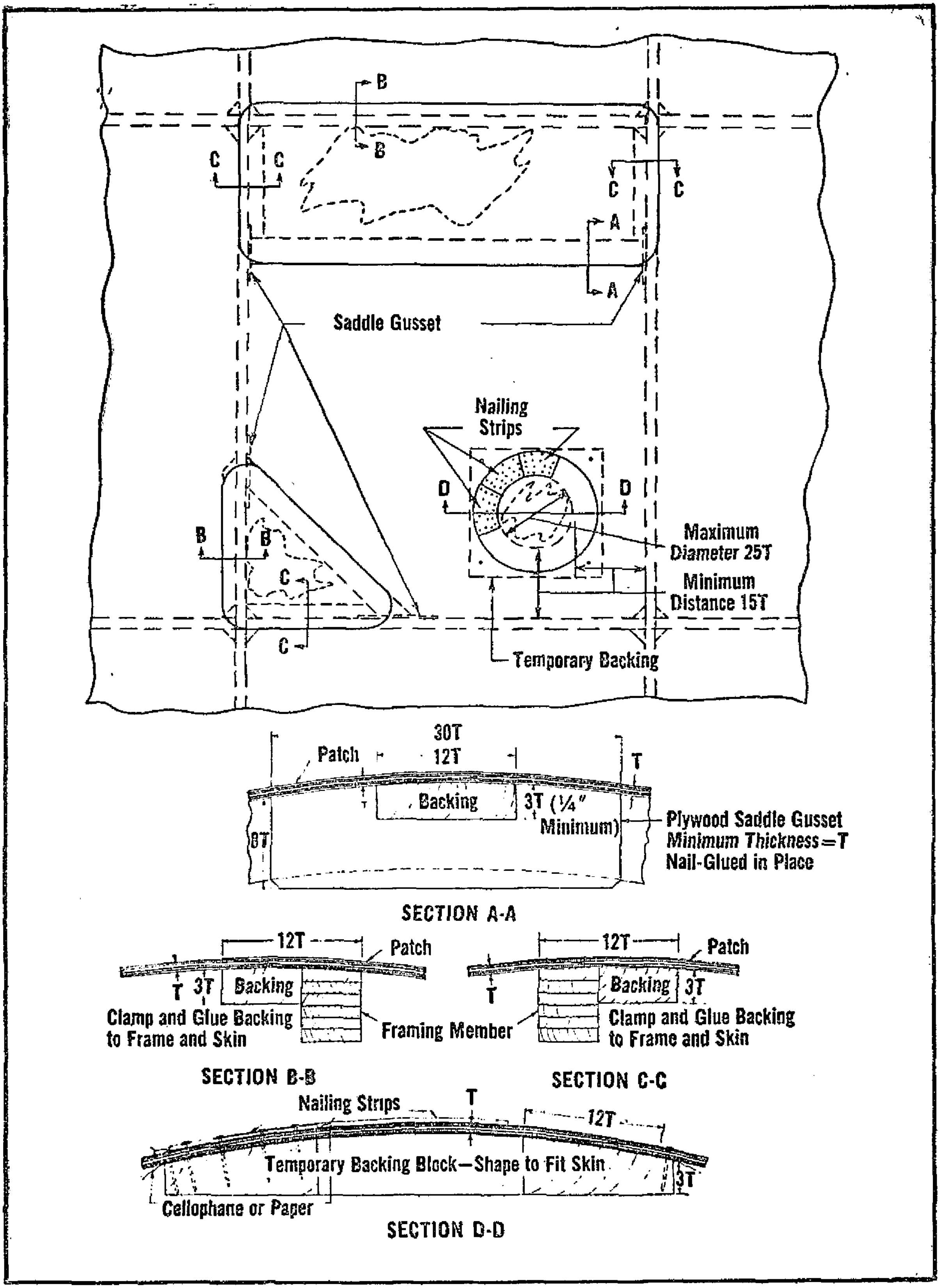


FIGURE 1-14. Scarf Patches—Back of Skin Accessible. Ref. CAM 18.20-1(e)(4)(i), 18.20-1(e)(4)(ii)

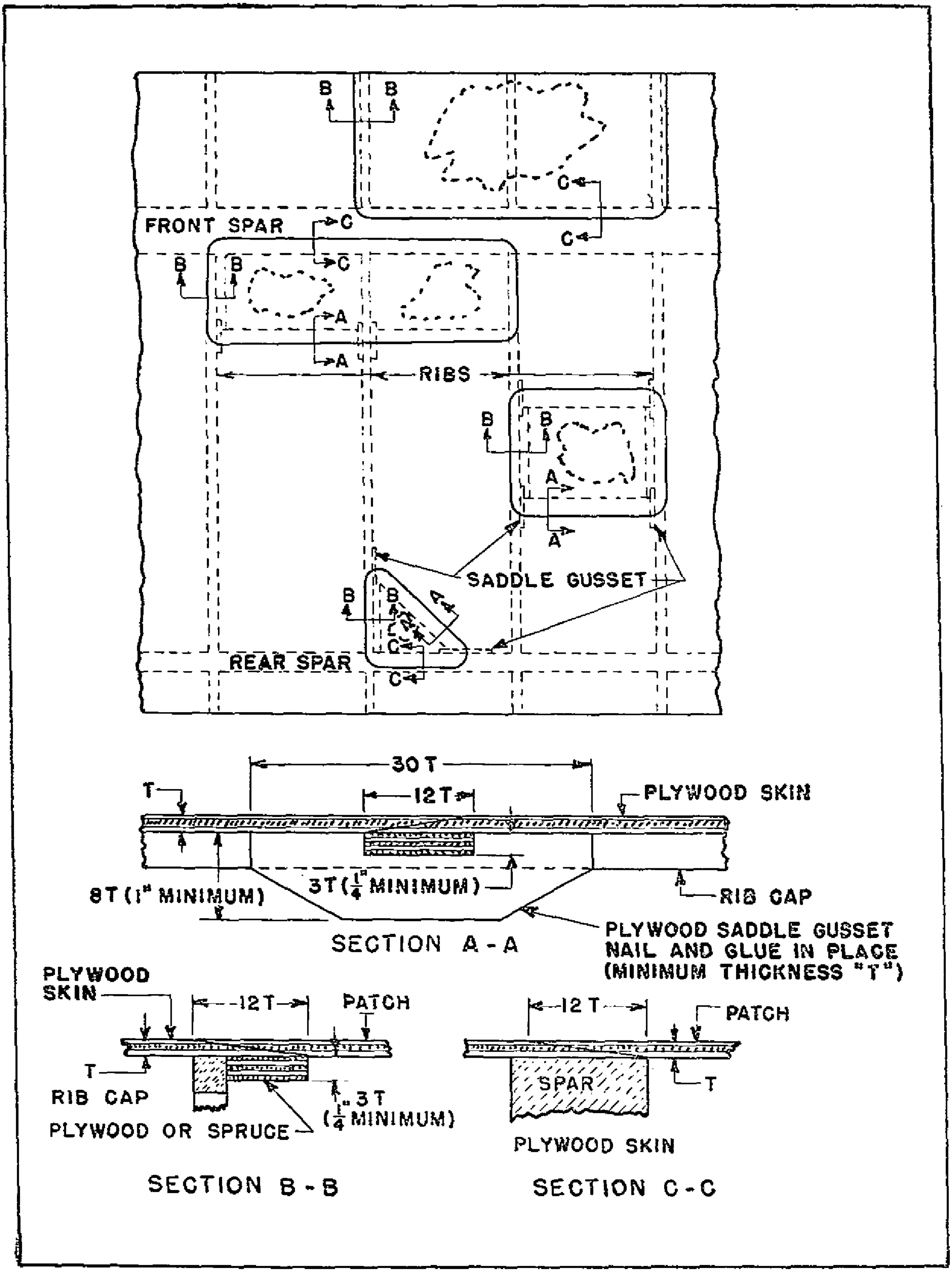
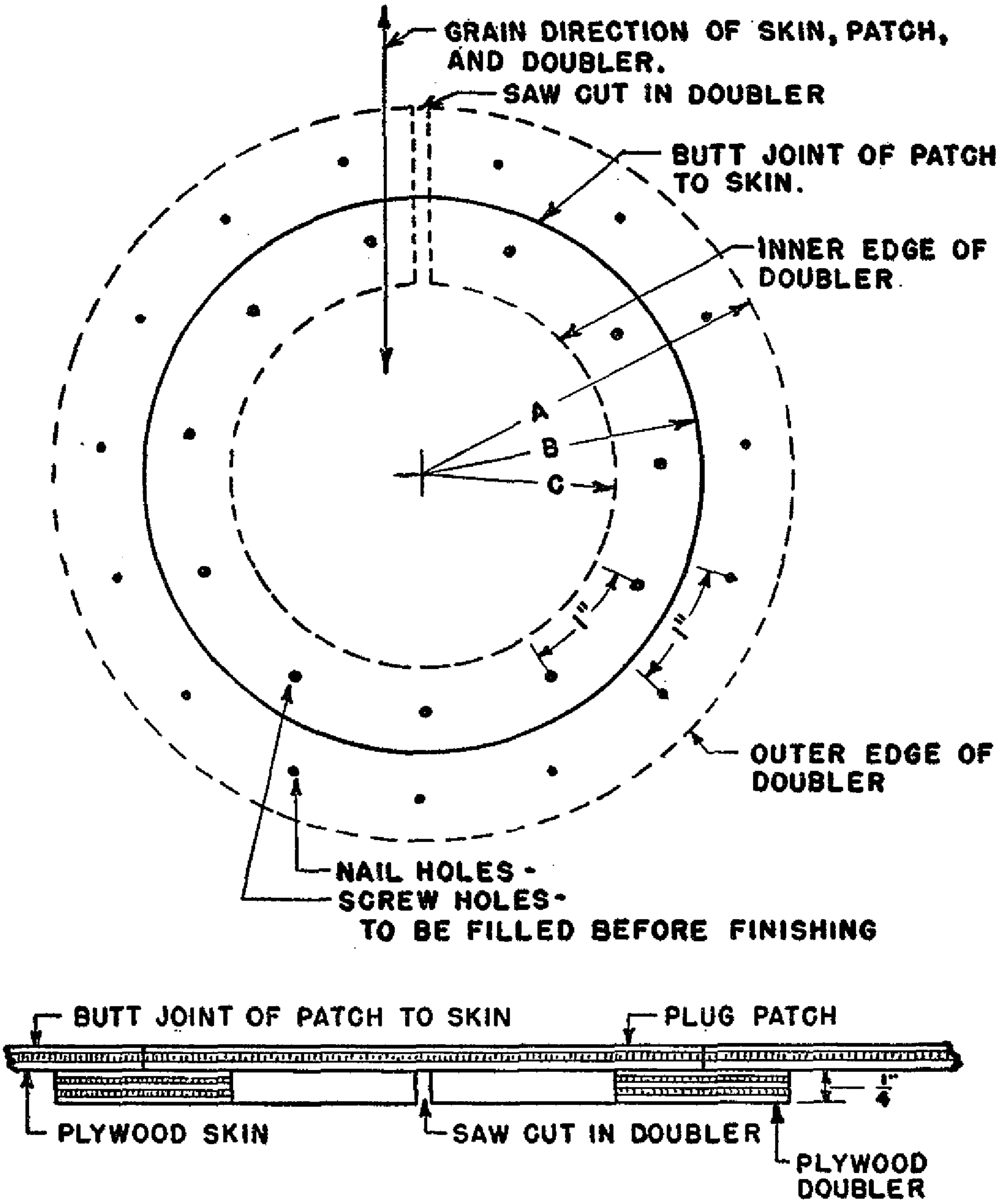


FIGURE 1–15. Scarf Patches—Back of Skin Inaccessible. Ref. CAM 18.20–1(e)(4)(i), 18.20–1(e)(4)(iii)(*a*)



DIMENSIONS

	A	B	C
SMALL CIRCULAR PLUG PATCH	$2\frac{5}{8}$	2	$1\frac{3}{8}$
LARGE CIRCULAR PLUG PATCH	$3\frac{7}{8}$	3	$2\frac{1}{8}$

(TWO ROWS OF SCREWS AND NAILS REQUIRED FOR LARGE PATCH.)

FIGURE 1–16. Round Plug-Patch Assembly. Ref. CAM 18.20–1(e)(5)(i), 18.20–1(e)(5)(iii)

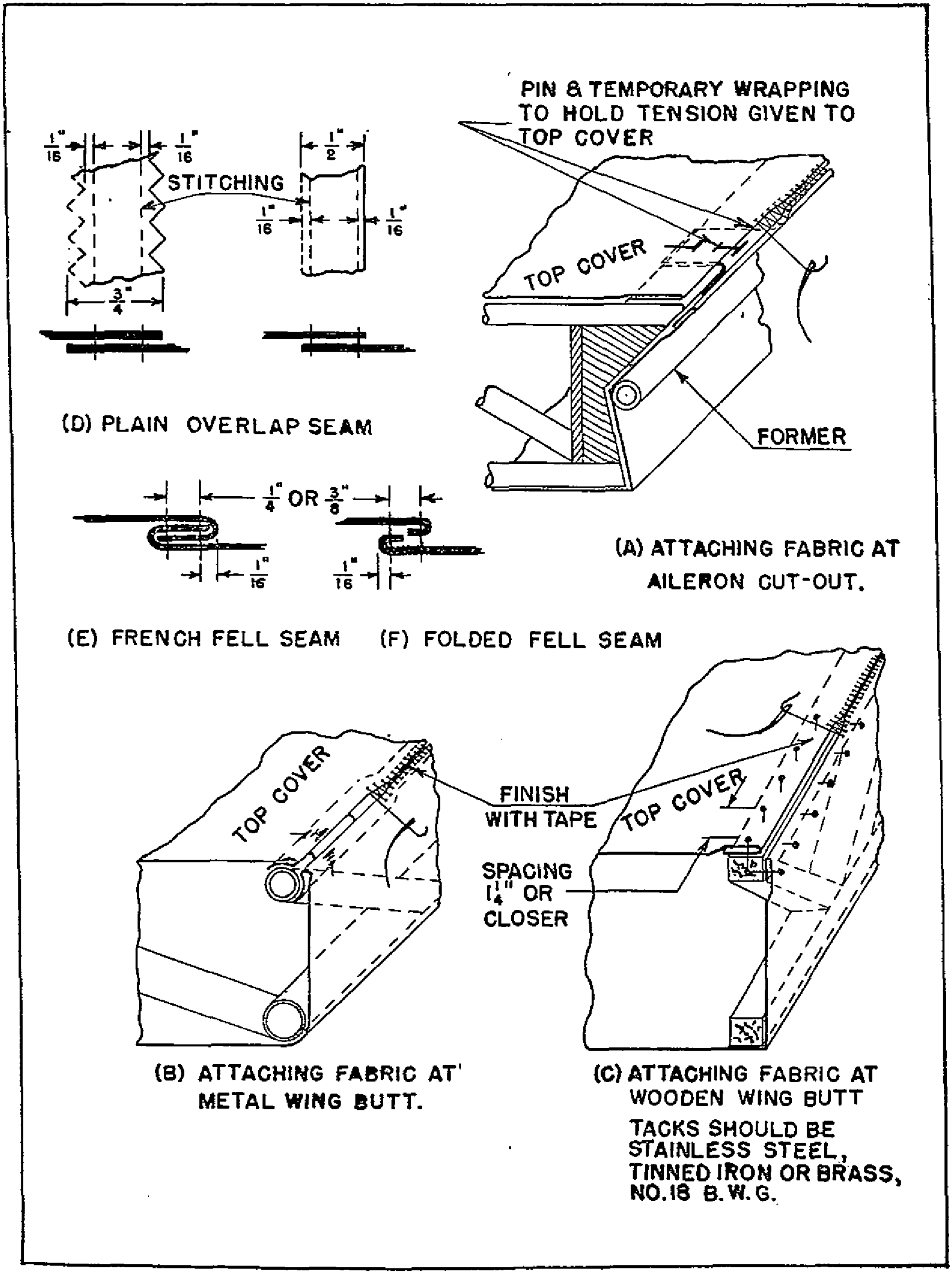


FIGURE 2-1. Typical Methods of Attaching Fabric. Ref. CAM 18.20-2(b)(4)(ii)(a)

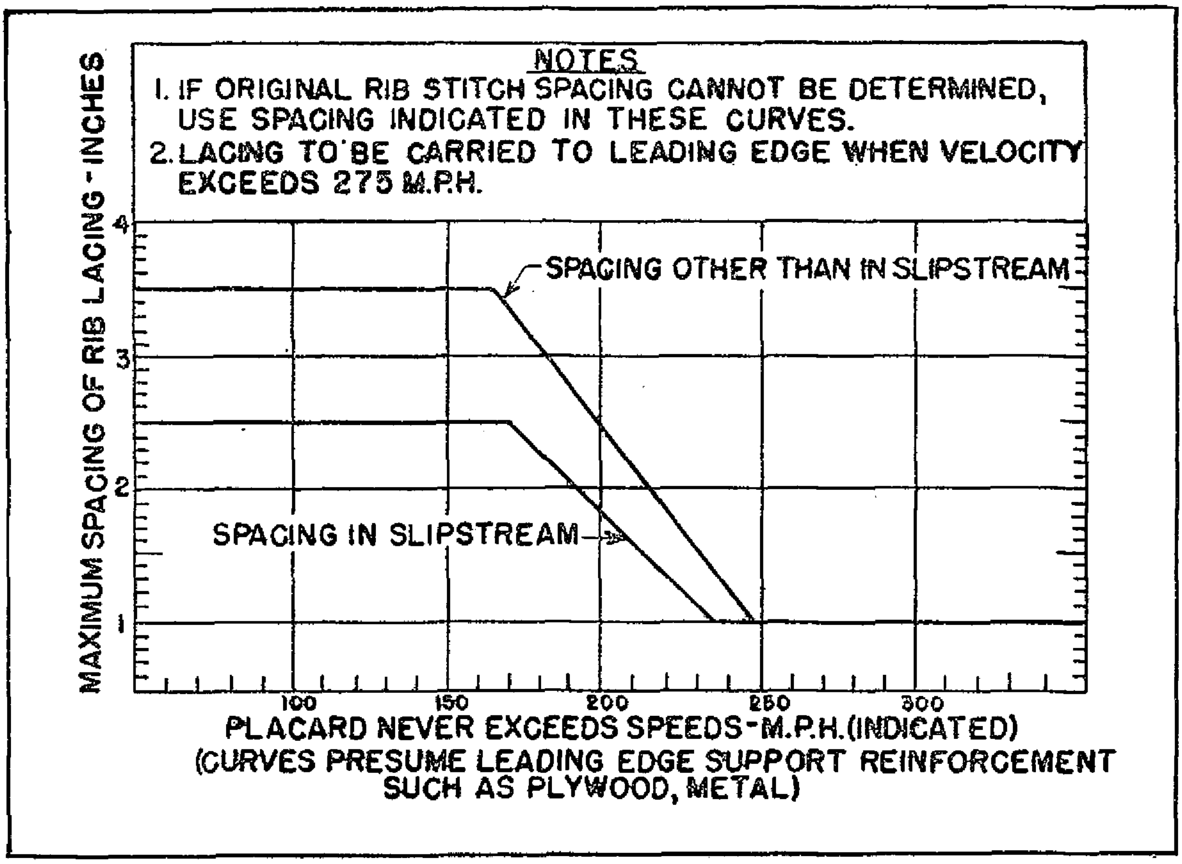


FIGURE 2-2. Fabric Attachment Spacing. Ref. CAM 18.20-2(b)(8)

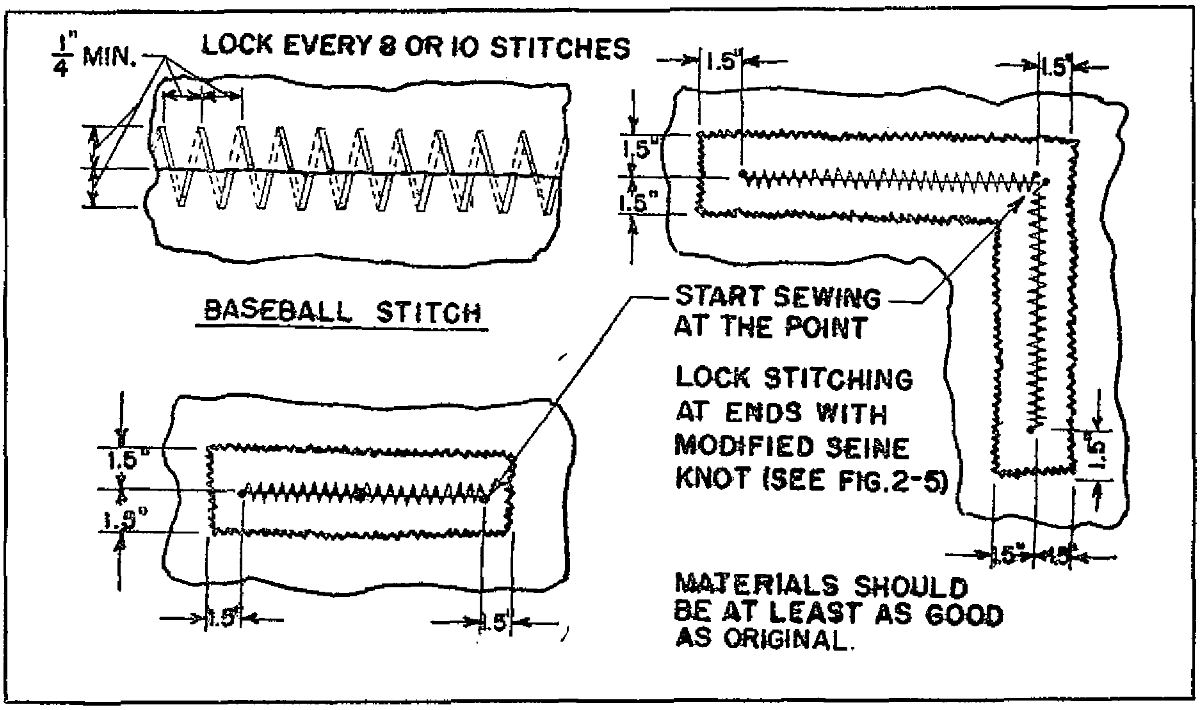


FIGURE 2-3. Repair of Tears in Fabric. Ref. CAM 18.20-2(d)(1)

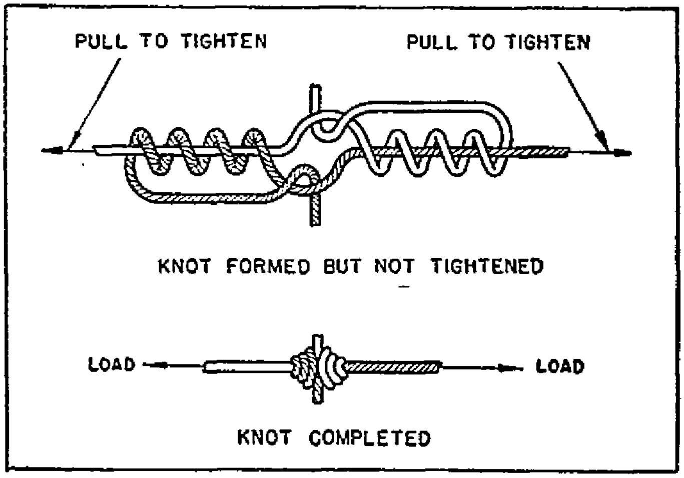


FIGURE 2-4. Splice Knot. Ref. CAM 18.20-2(b)(7)(i)

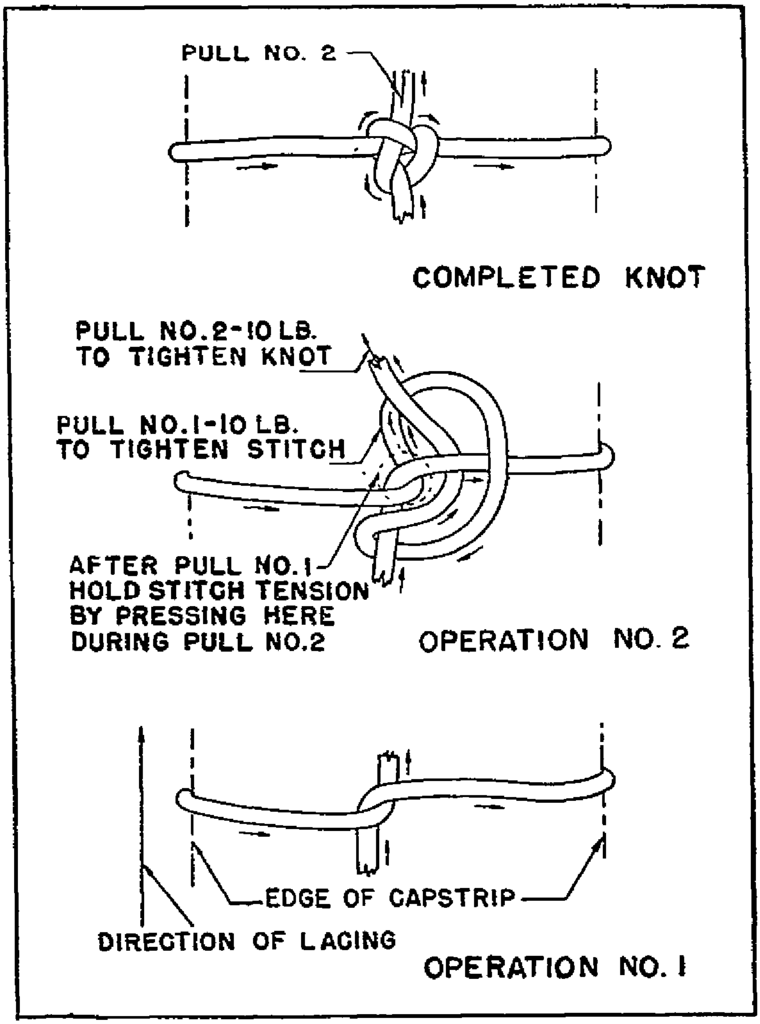


FIGURE 2-5. Standard Knot for Rib Lacing (Modified Seine Knot)
Ref. CAM 18.20-2(b)(4)(ii)(*b*), 18.20-2(b)(7)(i)

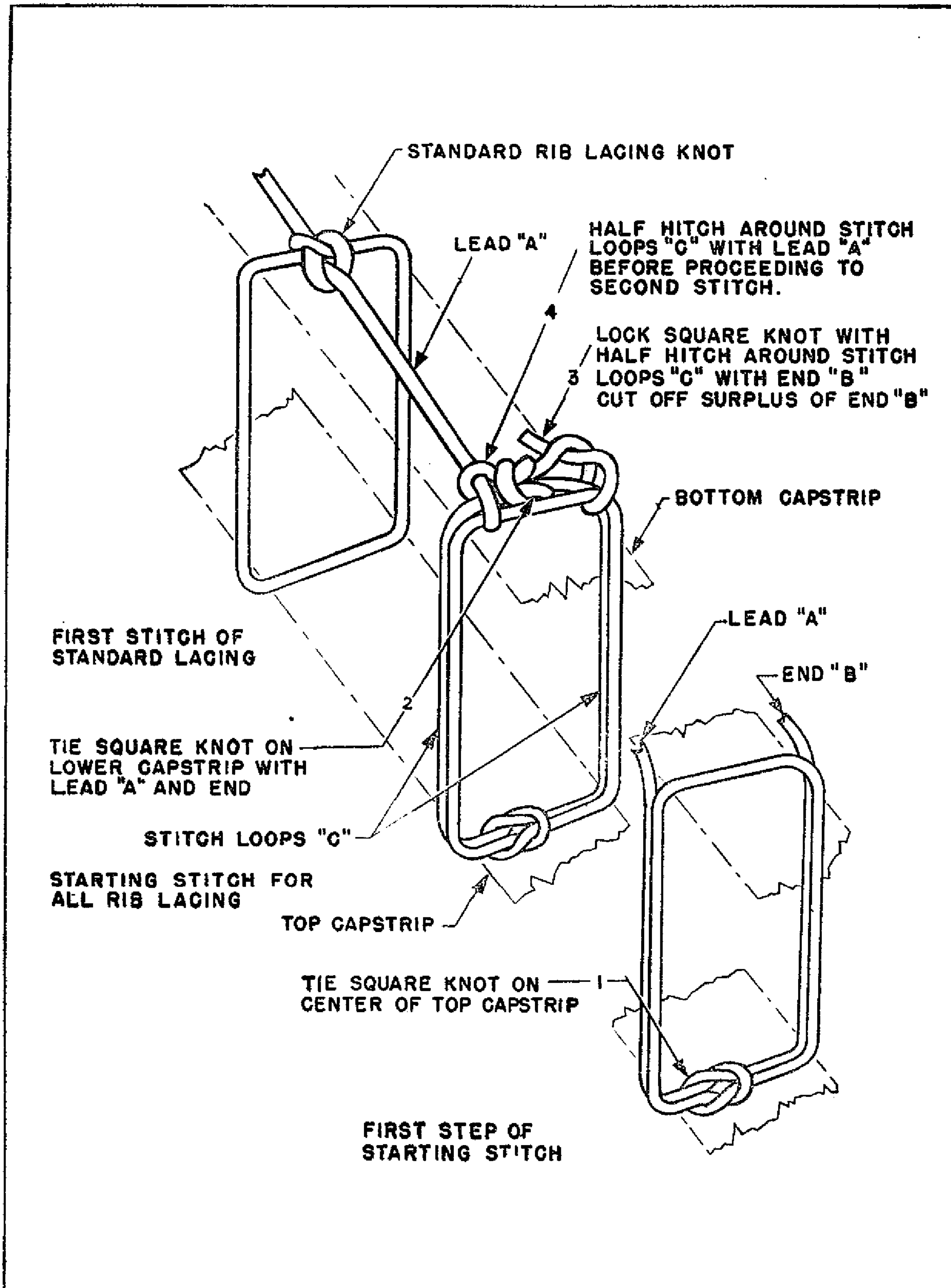


FIGURE 2-6. Starting Stitch for Rib Stitching. Ref. CAM 18.20-2(b)(7)(i)

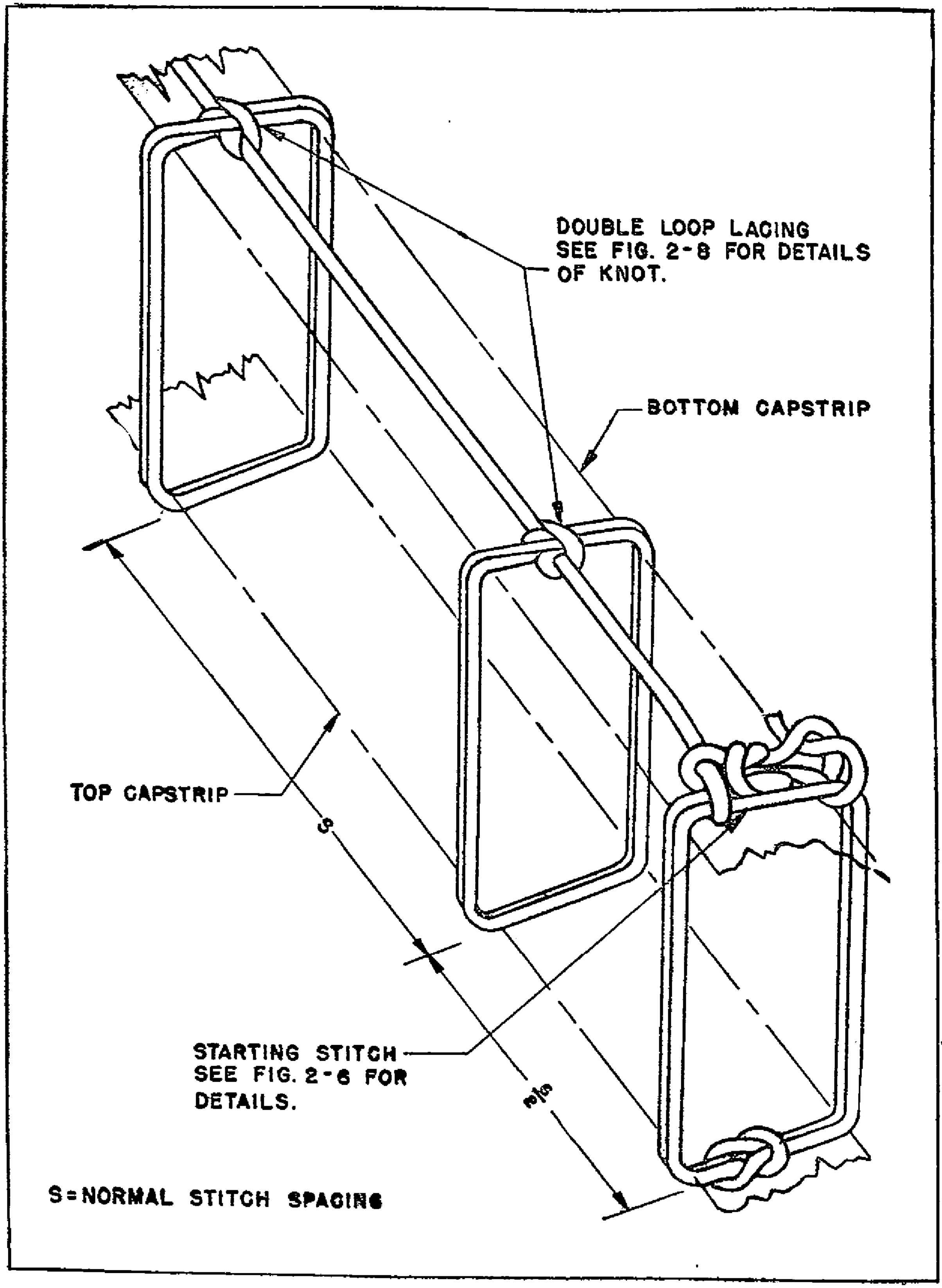


FIGURE 2-7. Standard Double-Loop Lacing. Ref. CAM 18.20-2(b)(7)(ii)

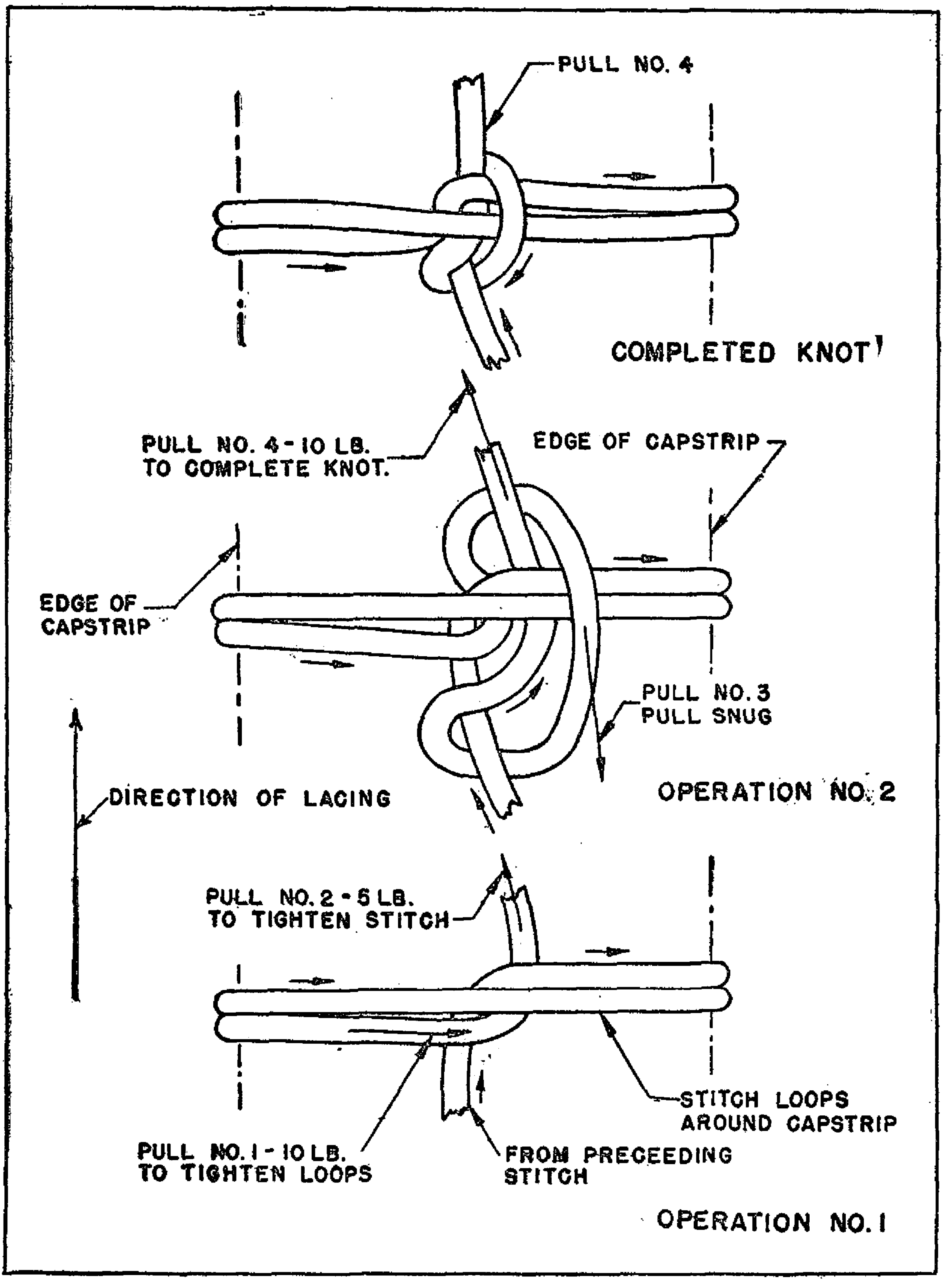


FIGURE 2-8. Standard Knot for Double-Loop Lacing. Ref. CAM 18.20-2(b)(7)(ii)

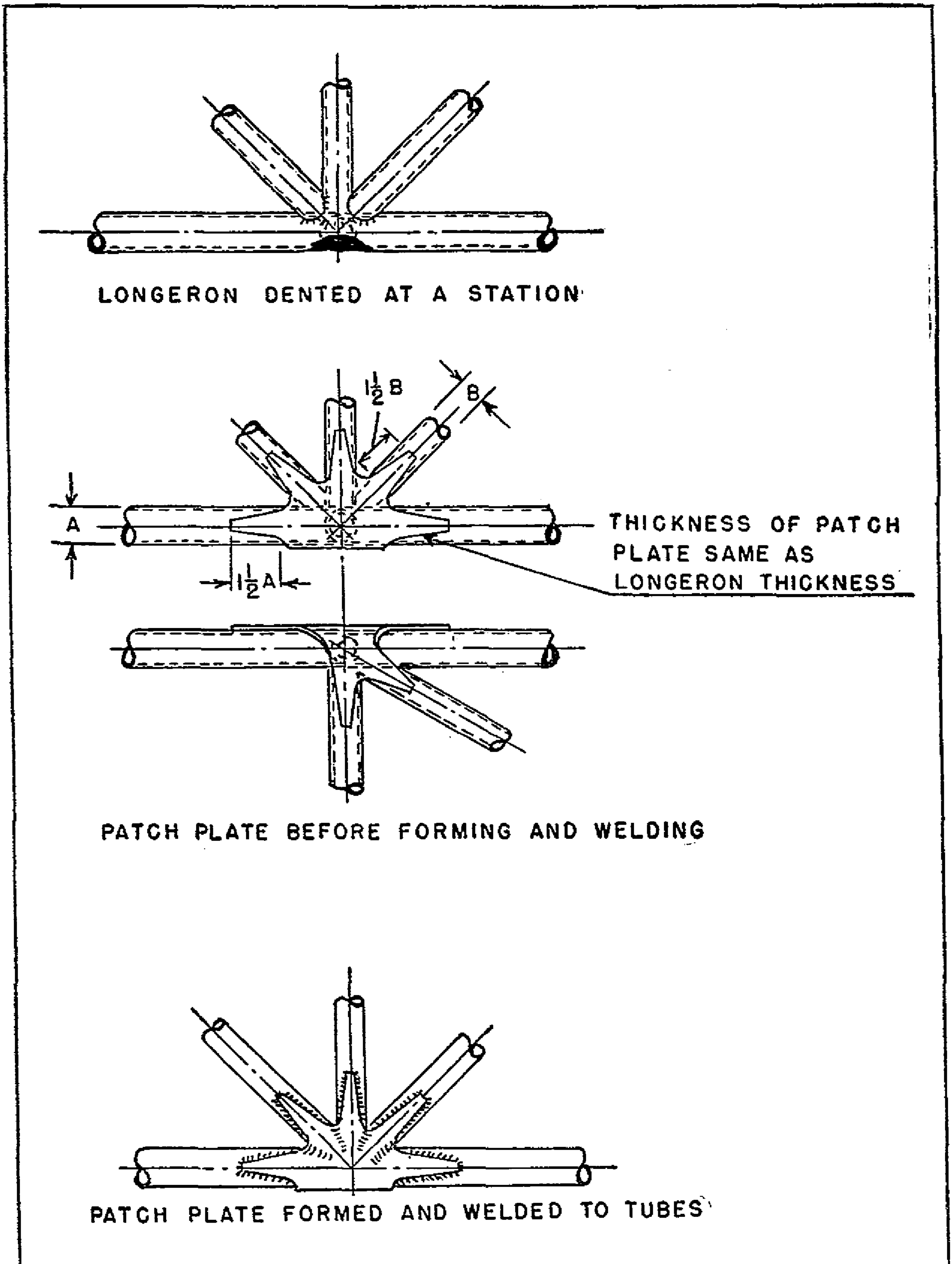


FIGURE 3–1. Members Dented at a Cluster. Ref. CAM 18.20–3 (b) (2) (ii)

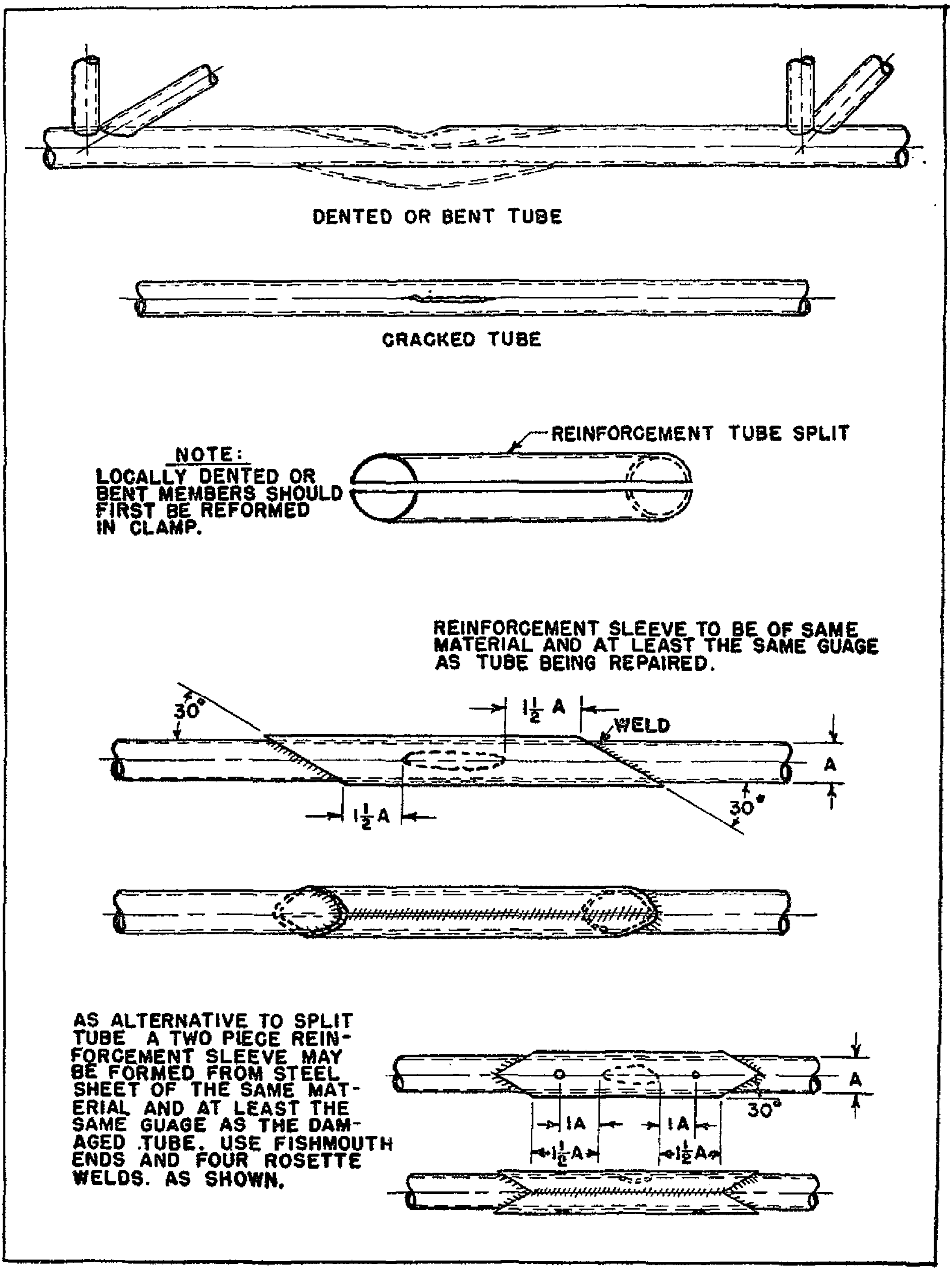


FIGURE 3-2. Members Dented in a Bay—Repair by Welded Sleeve. Ref. CAM 18.20-3(b)(2)(iii)(*a*), 18.20-3(b)(5)(i), 18.20-3(b)(5)(iv), 18.20-3(b)(7)

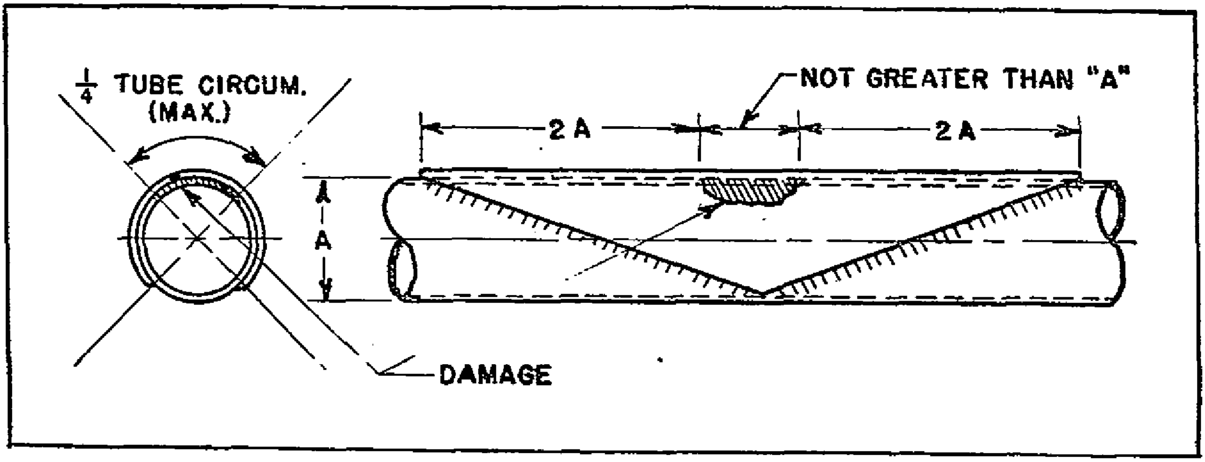


FIGURE 3-3. Welded Patch Repair. Ref. CAM 18.20-3(b)(2)(iv), 18.20-3(b)(5)(iv)

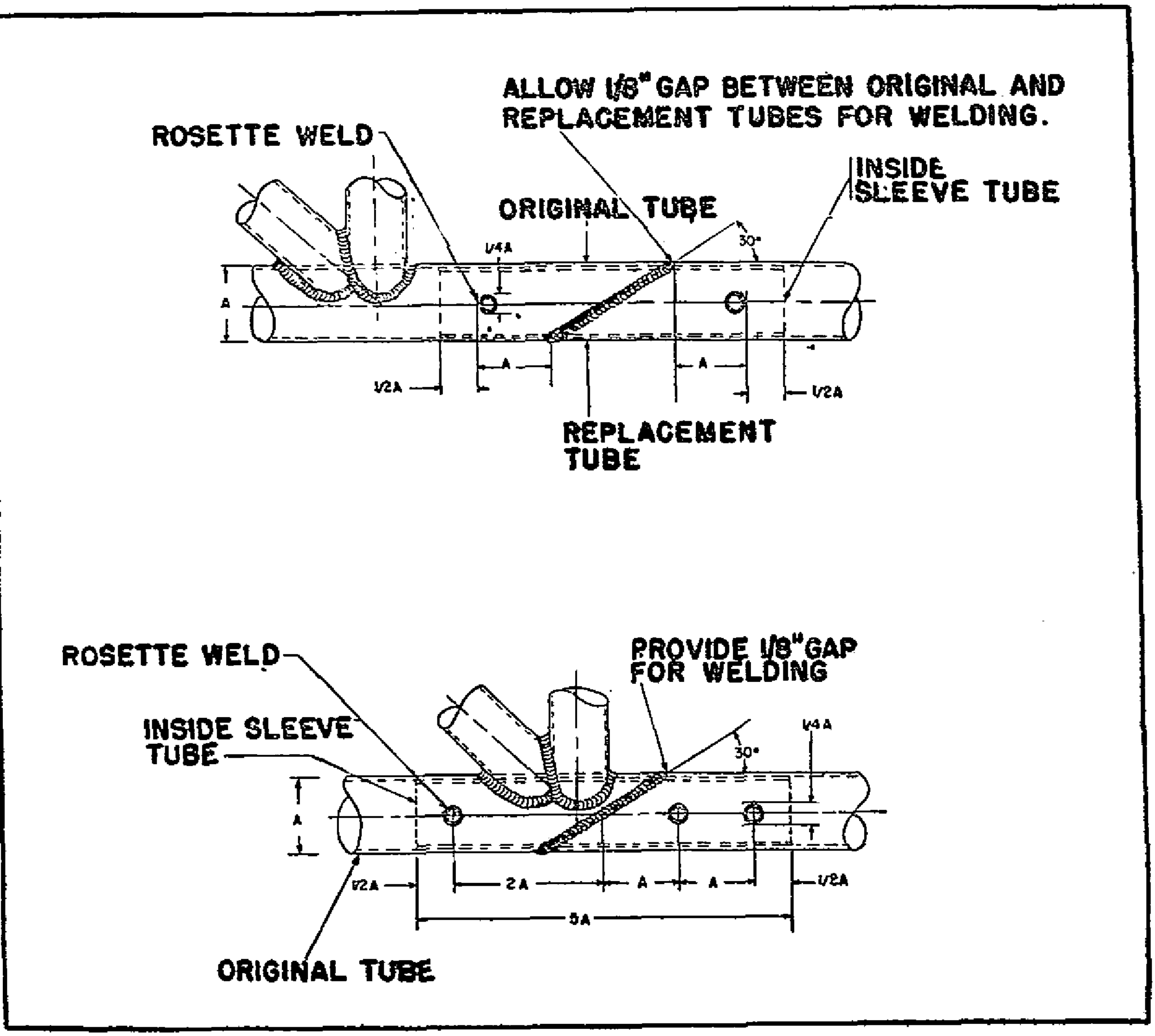


FIGURE 3-4. Splicing by Inner Sleeve Method. Ref. CAM 18.20-3(b)(2)(v), 18.20-3(b)(5)(iv), 18.20-3(b)(7)

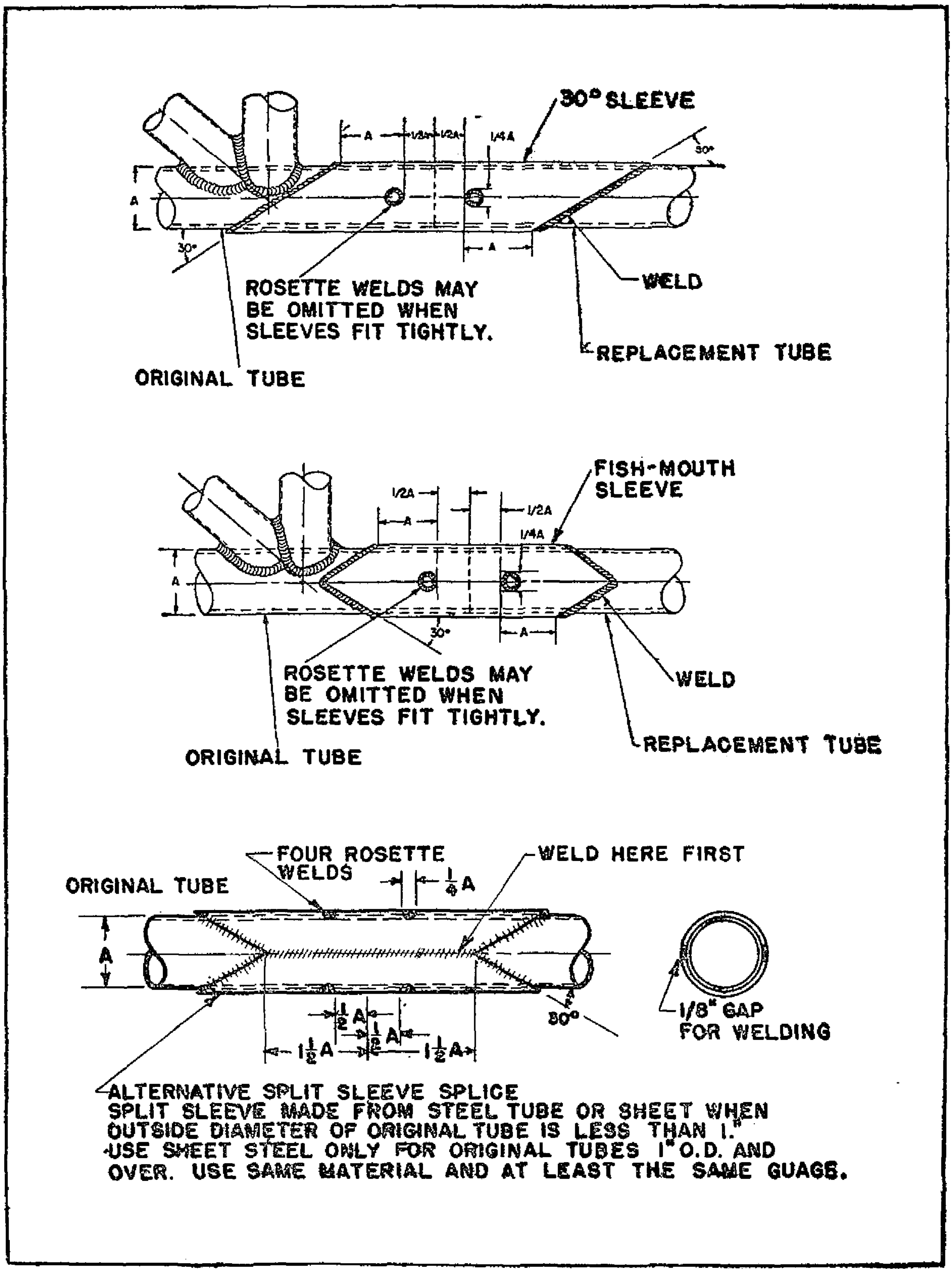


FIGURE 3-5. Splicing by Outer Sleeve Method—Replacement by Welded Outside Sleeve.
Ref. CAM 18.20-3(b)(2)(vi), 18.20-3(b)(5)(iv), 18.20-3(b)(7)

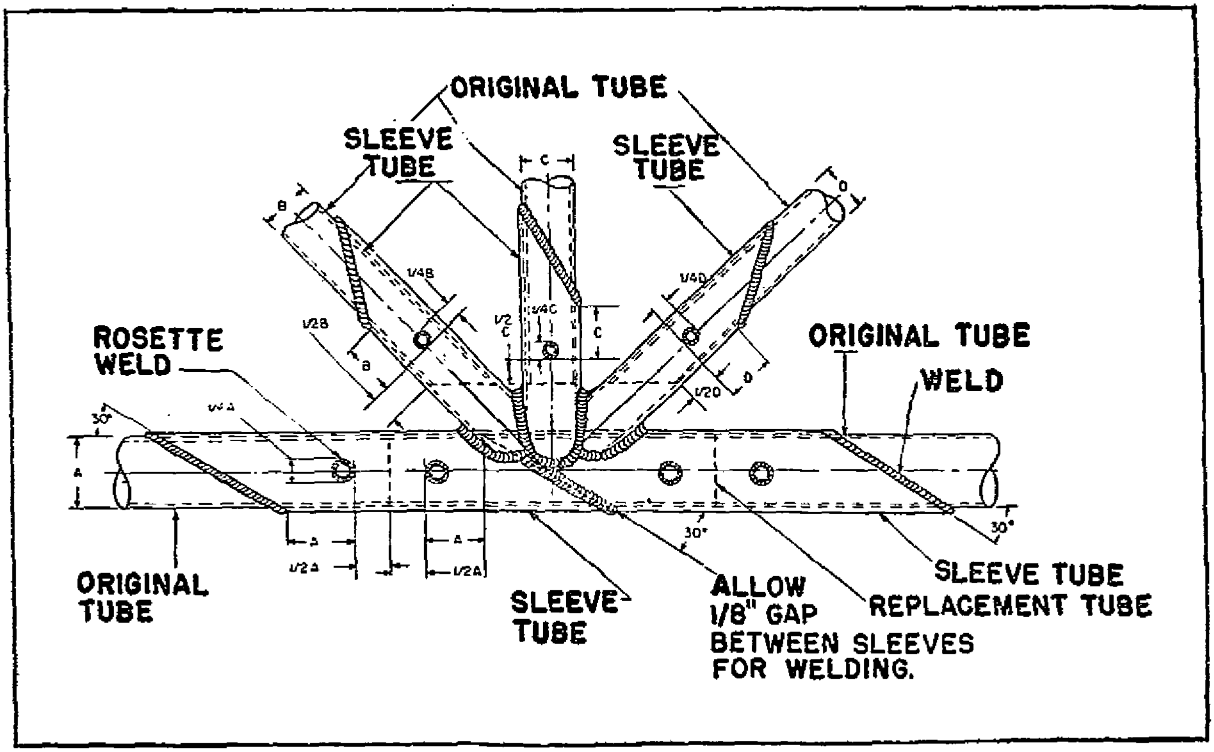


FIGURE 3-6. Tube Replacement at a Station by Welded Outer Sleeves. Ref. CAM 18.20-3(b)(2)(vi), 18.20-3(b)(5)(iv)

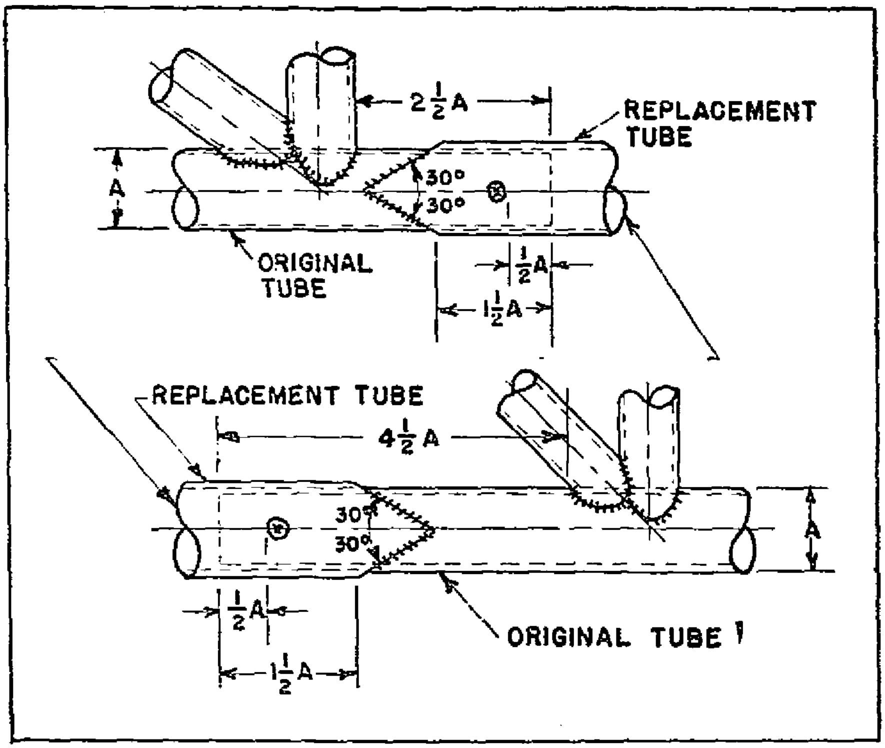


FIGURE 3-7. Splicing Using Larger Diameter Replacement Tube. Ref. CAM 18.20-3(b)(2)(vii), 18.20-3(b)(5)(iv)

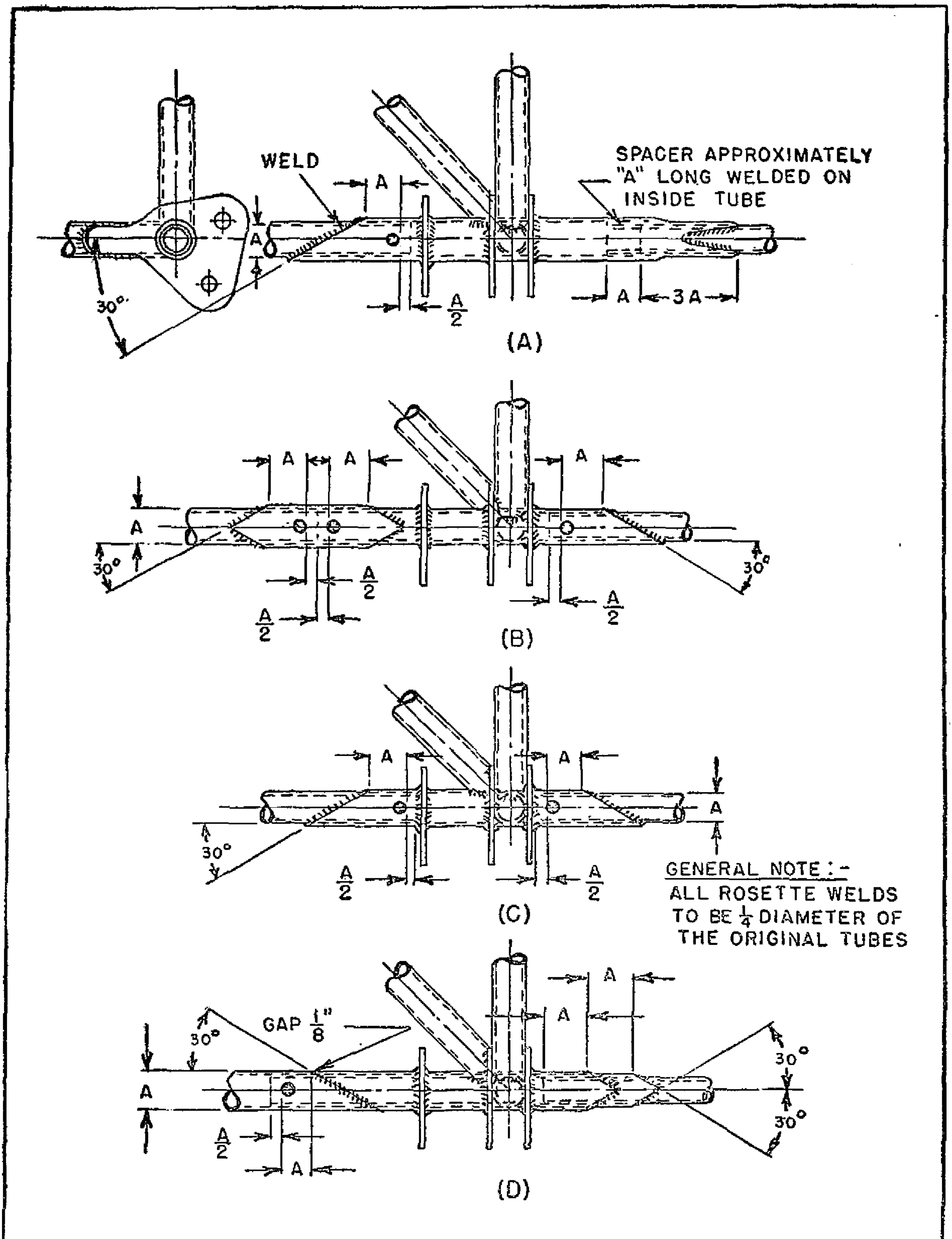
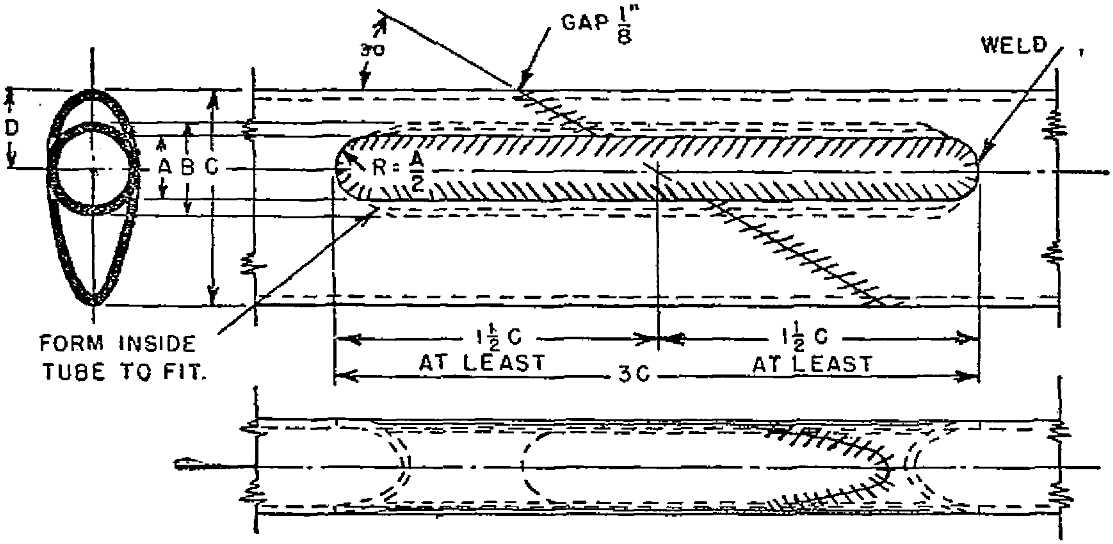


FIGURE 3-8. Repairs at Built-in Fuselage Fittings. Ref. CAM 18.20-3(b)(3), 18.20-3(b)(5)(i), 18.20-3(b)(5)(iv), 18.20-3(c)(6)(i)(c)



A - SLOT WIDTH (ORIGINAL TUBE)

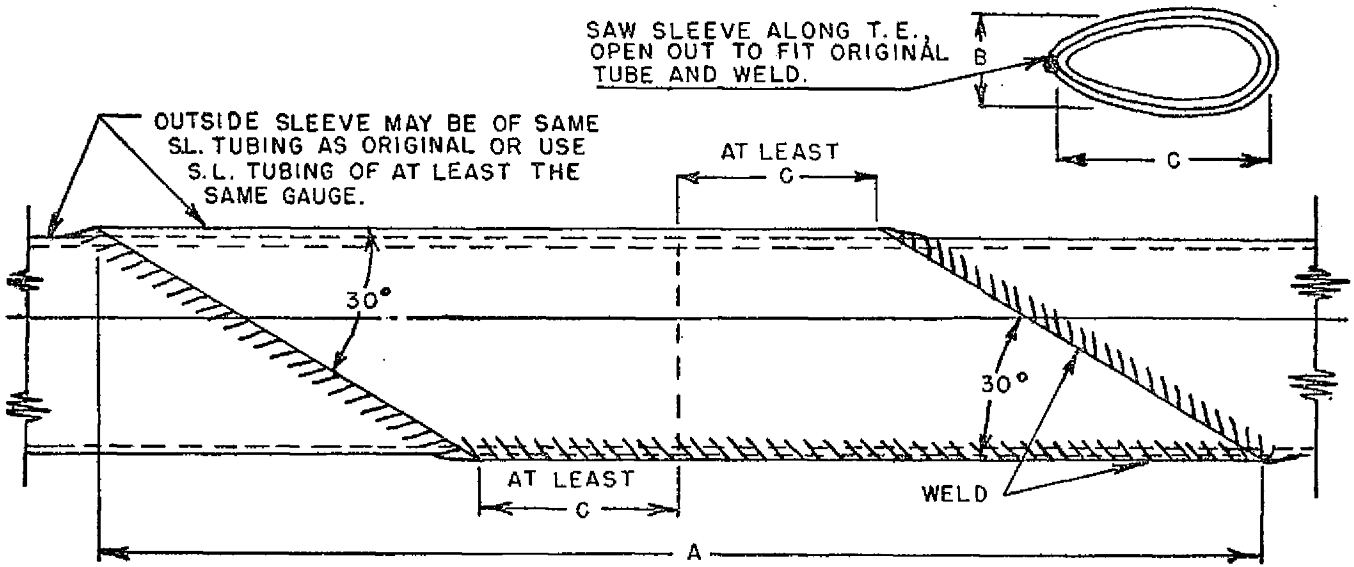
B - OUTSIDE DIAMETER (INSERT TUBE)

C - STREAMLINE TUBE LENGTH OF MAJOR AXIS

S.L. SIZE	A	B	C	D
1"	3/8"	9/16"	1.340"	.496"
1-1/4	3/8	11/16	1.670	.619
1-1/2	1/2	7/8	2.005	.743
1-3/4	1/2	1	2.339	.867
2	1/2	1-1/8	2.670	.991
2-1/4	1/2	1-1/4	3.008	1.115
2-1/2	1/2	1-3/8	3.342	1.239

ROUND INSERT TUBE (B) SHOULD BE AT LEAST OF SAME MATERIAL AND ONE GAUGE THICKER THAN ORIGINAL STREAMLINE TUBE (C).

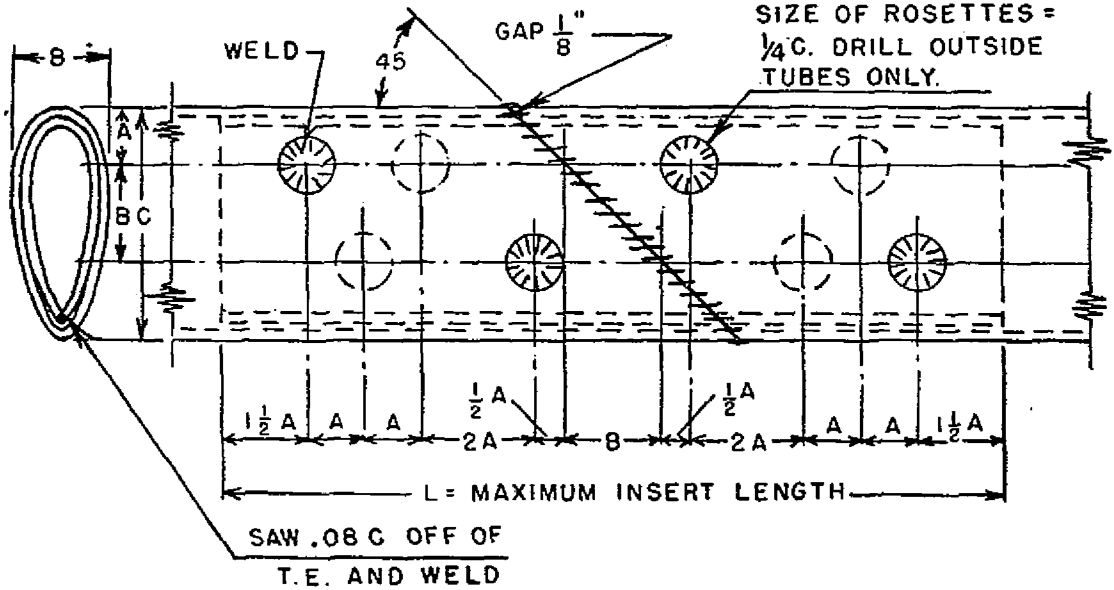
FIGURE 3-9. Streamline Tube Splice Using Round Tube (Applicable to Landing Gears.) Ref. CAM 18.20-3 (b) (3) (i), 18.20-3(b) (3) (iii), 18.20-3(b) (3) (iv), 18.20-3(b) (5) (ii), 18.20-3(b) (5) (iv)



A – MINIMUM LENGTH OF SLEEVE
B – STREAMLINE TUBE LENGTH OF MINOR AXIS
C – STREAMLINE TUBE LENGTH OF MAJOR AXIS

S. L. SIZE	A	B	C
1"	7.324"	.572"	1.340"
1-1/4	9.128	.714	1.670
1-1/2	10.960	.858	2.005
1-3/4	12.784	1.000	2.339
2	14.594	1.144	2.670
2-1/4	16.442	1.286	3.008
2-1/2	18.268	1.430	3.342

FIGURE 3-10. Streamline Tube Splice Using Split Sleeve (Applicable to Wing and Tail Surface Brace Struts and Other Members). Ref. CAM 18.20-3(b) (5) (iv), 18.20-3(b) (7)



INSERT TUBE IS OF SAME STREAMLINE TUBING AS ORIGINAL.

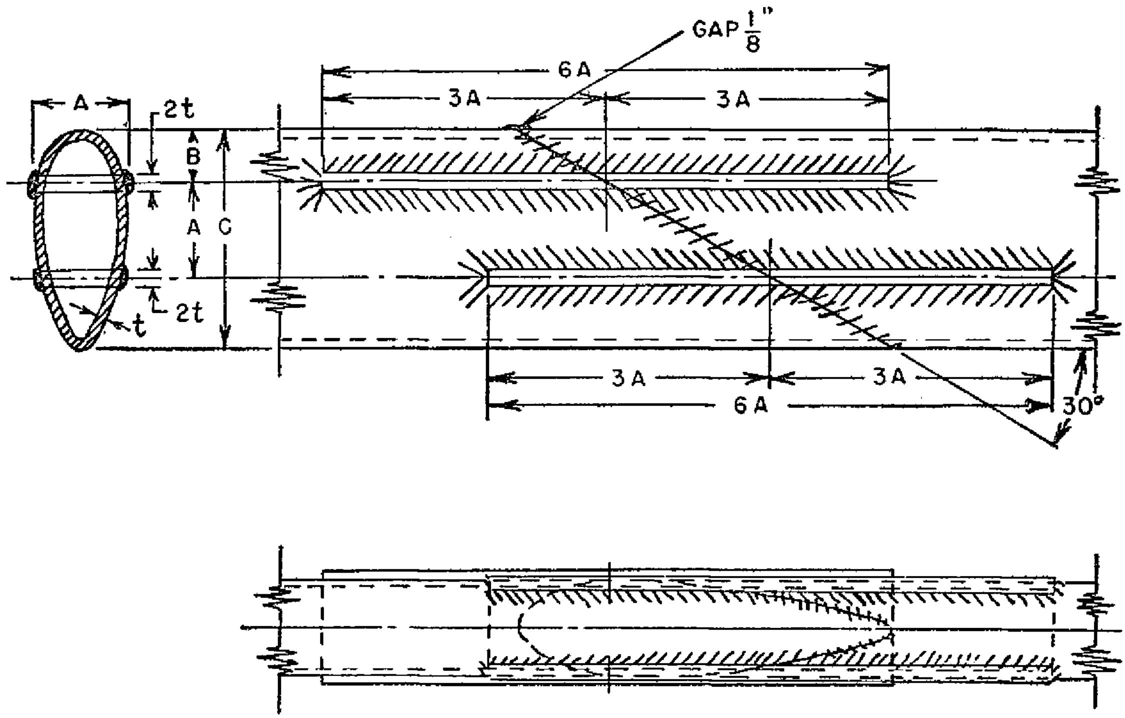
A IS $\frac{2}{3}$ B

B IS MINOR AXIS LENGTH OF ORIGINAL STREAMLINE TUBE

C IS MAJOR AXIS LENGTH OF ORIGINAL STREAMLINE TUBE

S. L. SIZE	A	B	C	L
1"	.382	.572	1.340	5.16
1 1/4	.476	.714	1.670	6.43
1 1/2	.572	.858	2.005	7.72
1 3/4	.667	1.000	2.339	9.00
2	.763	1.144	2.670	10.30
2 1/4	.858	1.286	3.008	11.58
2 1/2	.954	1.430	3.342	12.88

FIGURE 3–11. Streamline Tube Splice Using Split Insert (Applicable to Landing Gears).
Ref. CAM 19.20–3(b)(5)(iv)



A- STREAMLINE TUBE LENGTH OF MINOR AXIS, PLATE WIDTHS.
B- DISTANCE OF FIRST PLATE FROM LEADING EDGE, $\frac{2}{3}$ A.
C- STREAMLINE TUBE LENGTH OF MAJOR AXIS.

S.L. SIZE	A	B	C	6A
1"	.572	.382	1.340	3.43
1-1/4	.714	.476	1.670	4.28
1-1/2	.858	.572	2.005	5.15
1-3/4	1.000	.667	2.339	6.00
2	1.144	.762	2.670	6.86
2-1/4	1.286	.858	3.008	7.72
2-1/2	1.430	.954	3.342	8.58

FIGURE 3-12. Streamline Tube Splice Using Plates (Applicable to Landing Gears). Ref. CAM 18.20-3(b)(5)(ii), 18.20-3(b)(5)(iv)

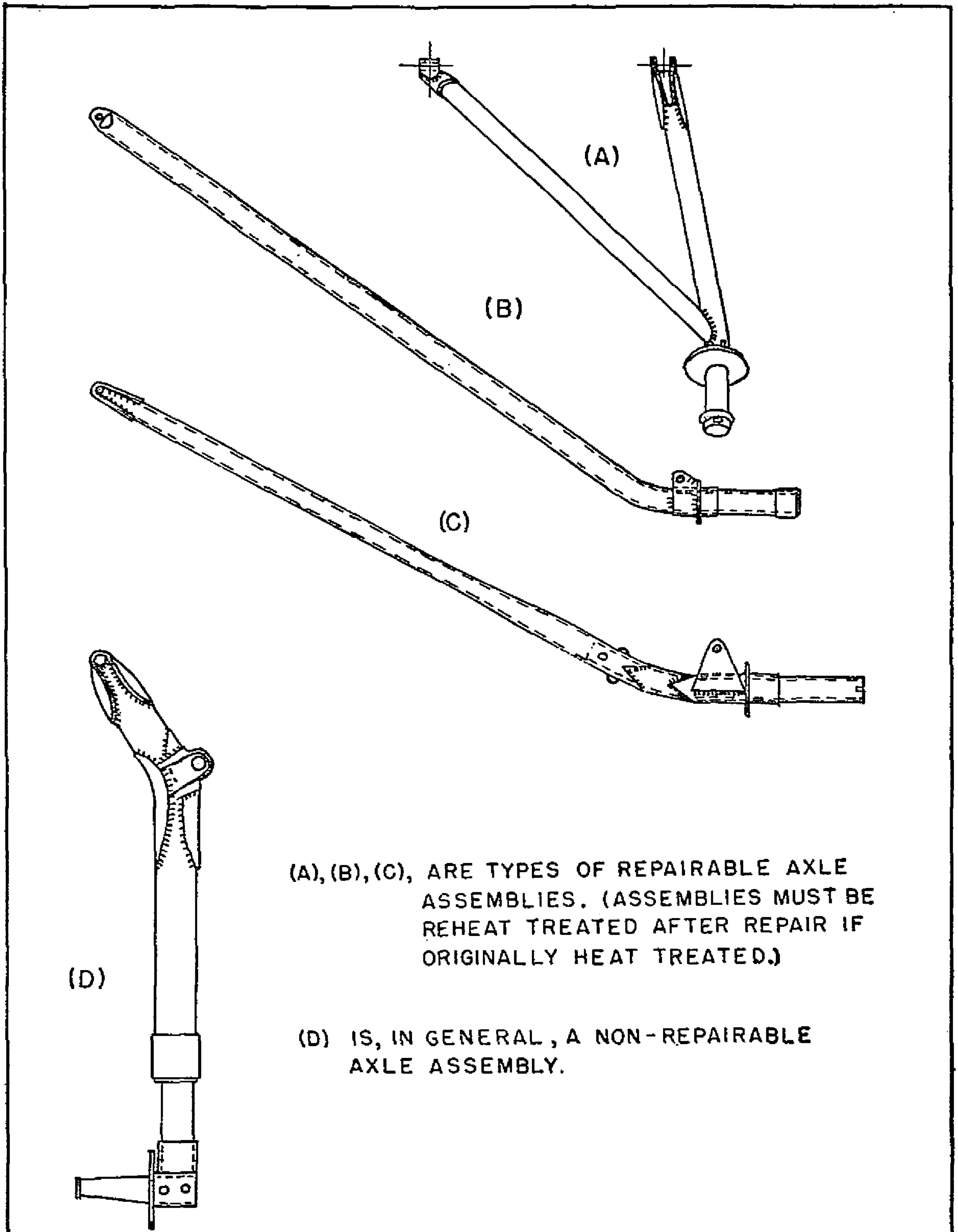


FIGURE 3-13. Representative Types of Repairable and Nonrepairable Axle Assemblies.
Ref. CAM 18.20–3 (b) (5) (iii)

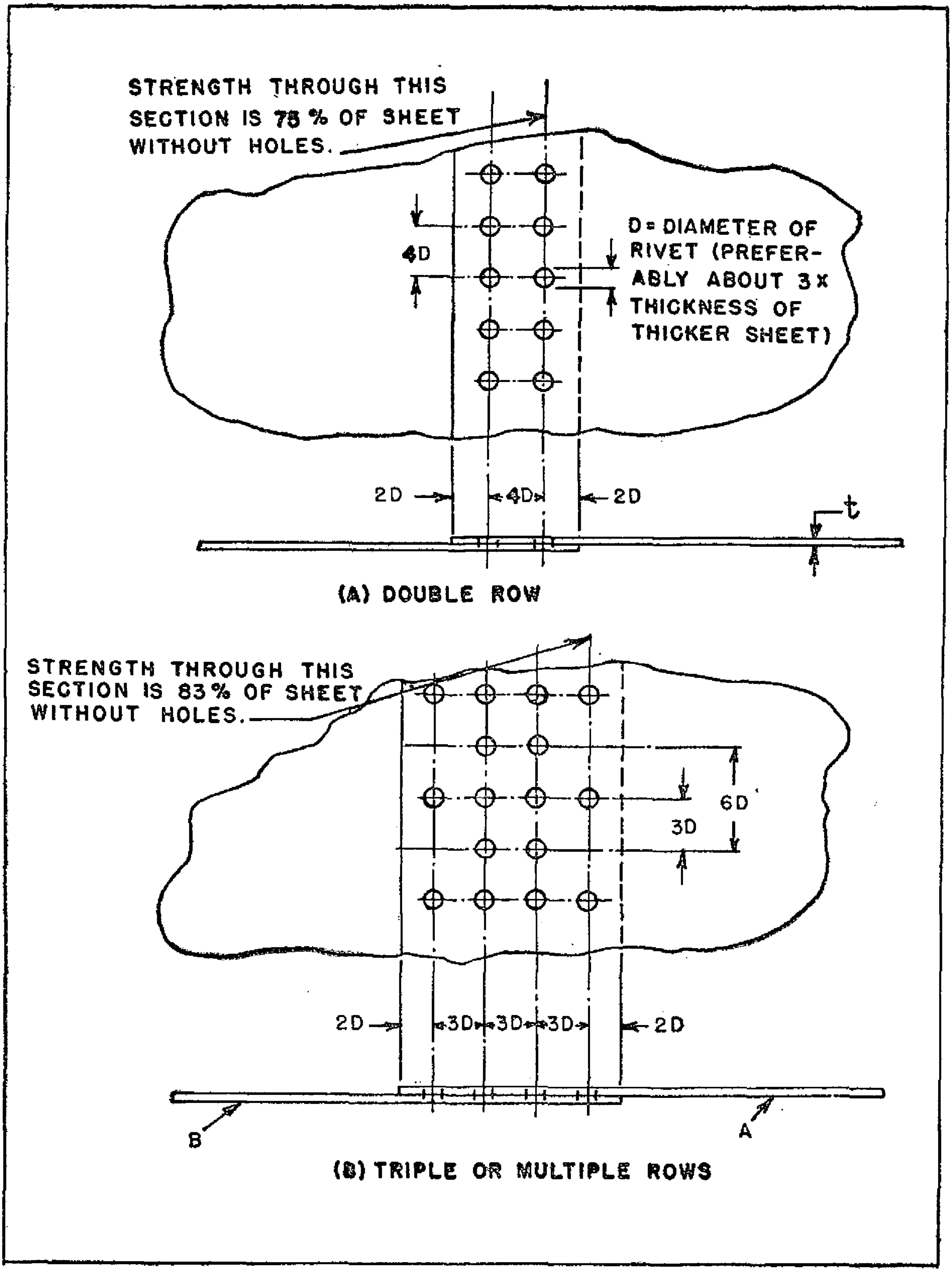


FIGURE 3-14. Rivet Hole Spacing and Edge Distance for Single-Lap Sheet Splices.
Ref. CAM 18.20-3(e)(4)(ii)(b), 18.20-3(e)(5)(vi)

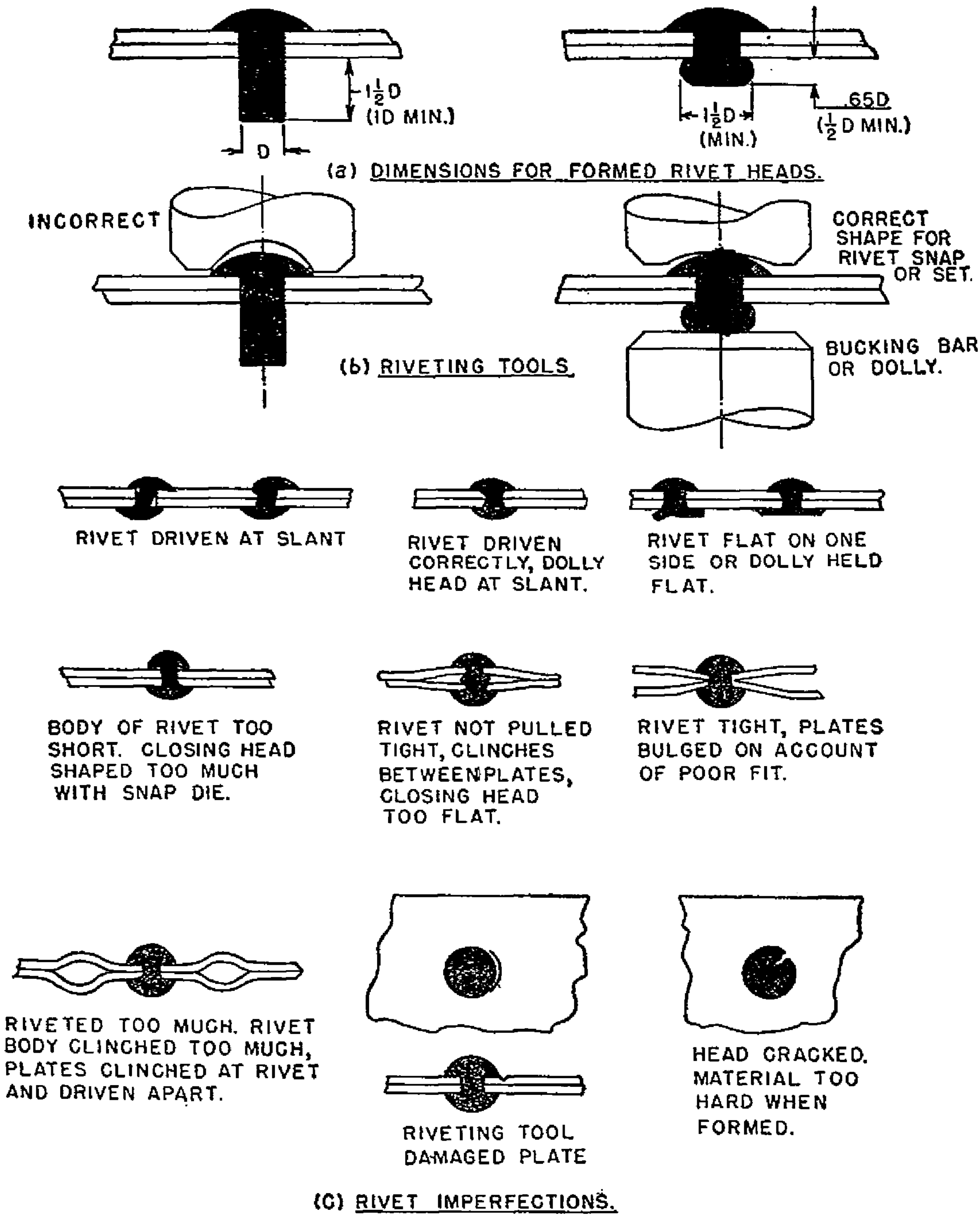
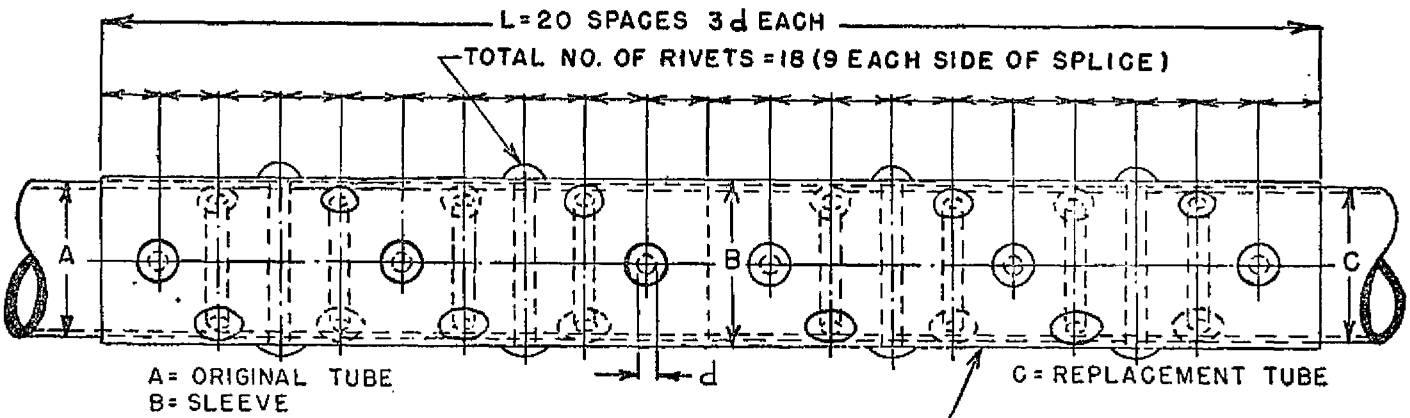
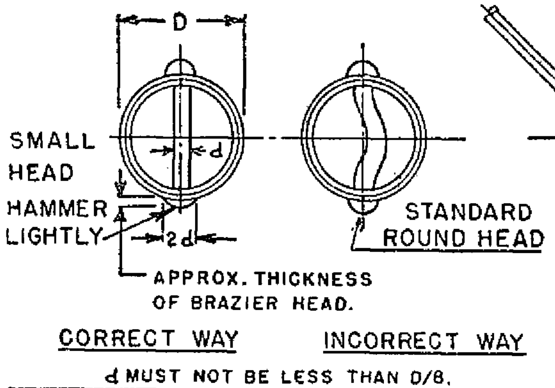
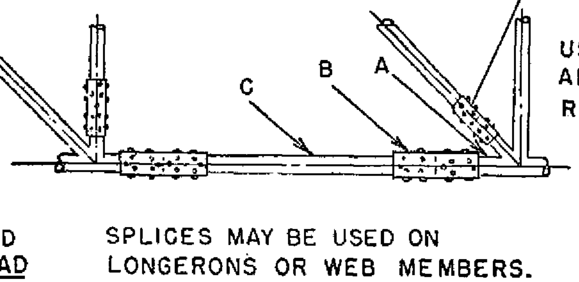
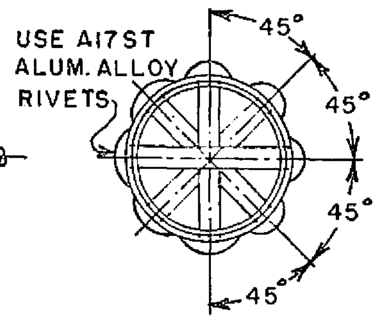


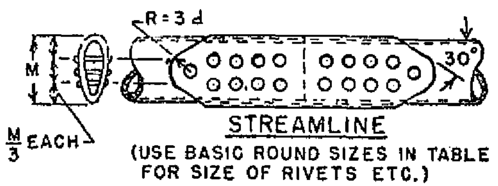
FIGURE 3–15. Riveting Practice and Rivet Imperfections. Ref. CAM 18.20–3(e)(4)(iv), 18.20–3(e)(5)(i)











A, C *	3/4	7/8	1	1-1/8	1-1/4	1-3/8	1-1/2	1-5/8	1-3/4	1-7/8
	.065	.065	.065	.065	.058	.058	.058	.058	.058	.058
B	7/8	1	1-1/8	1-1/4	1-3/8	1-1/2	1-5/8	1-3/4	1-7/8	2
	ALL .058 THICK									
RIVET DIA.	5/32	5/32	3/16	3/16	3/16	3/16	1/4	1/4	1/4	1/4
L	9-3/8	9-3/8	11-1/4	11-1/4	11-1/4	11-1/4	15	15	15	15

* INCLUDES ALL THICKNESSES UP TO AND INCLUDING MAXIMUM SHOWN.

NOTE: USE SAME MATERIAL AS ORIGINAL FOR SLEEVE AND REPLACEMENT TUBE.

FIGURE 3-16. Typical Repair Method for Tubular Members of Aluminum Alloy. Ref. CAM 18.20-3 (e) (5) (ii)

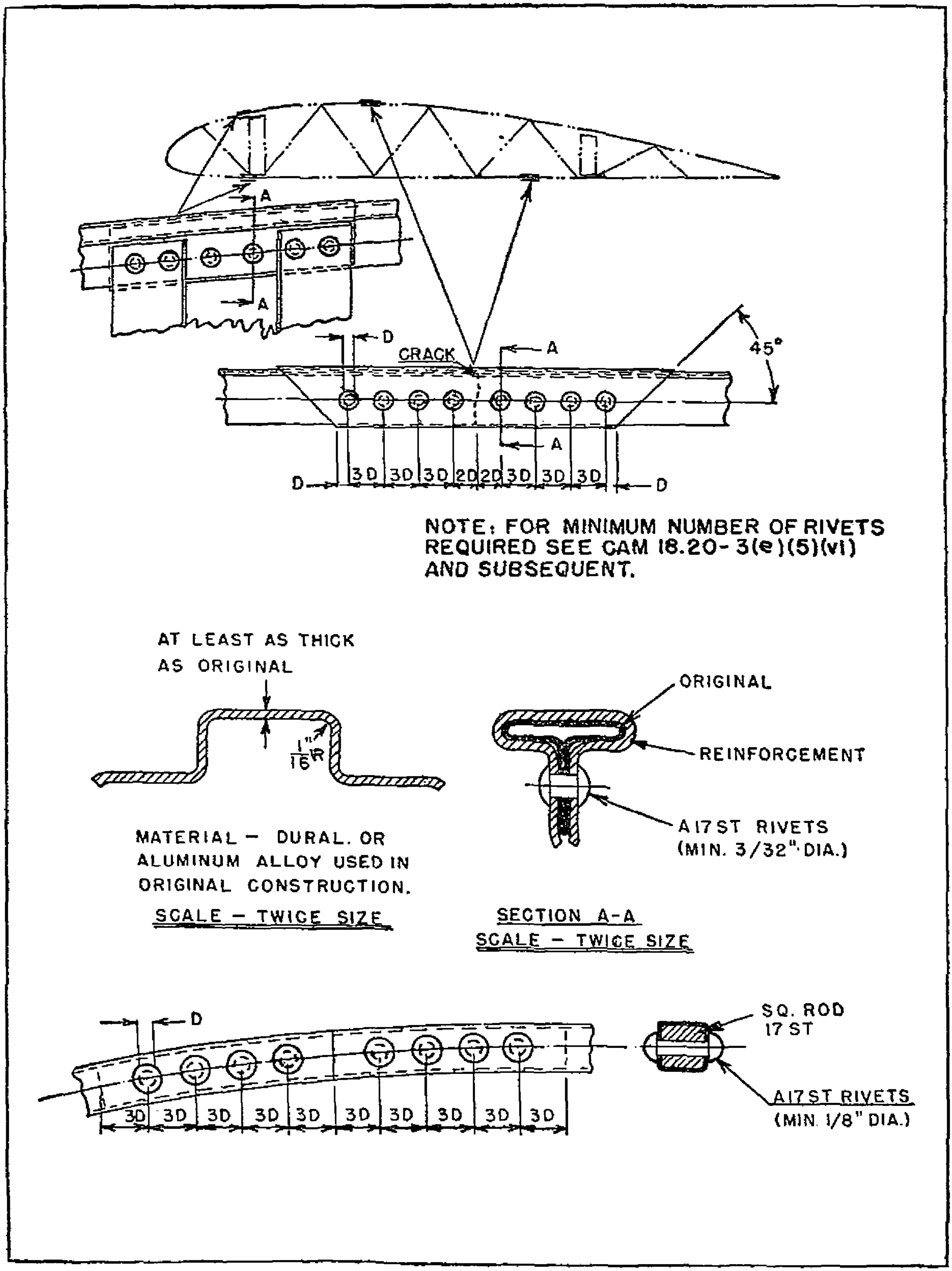


FIGURE 3-17. Typical Repair for Buckled or Cracked Formed Metal Wing Rib Capstrips Usually Found on Small or Medium Size Aircraft. Ref. *CAM 18.20-3(e)(5)(iv)*

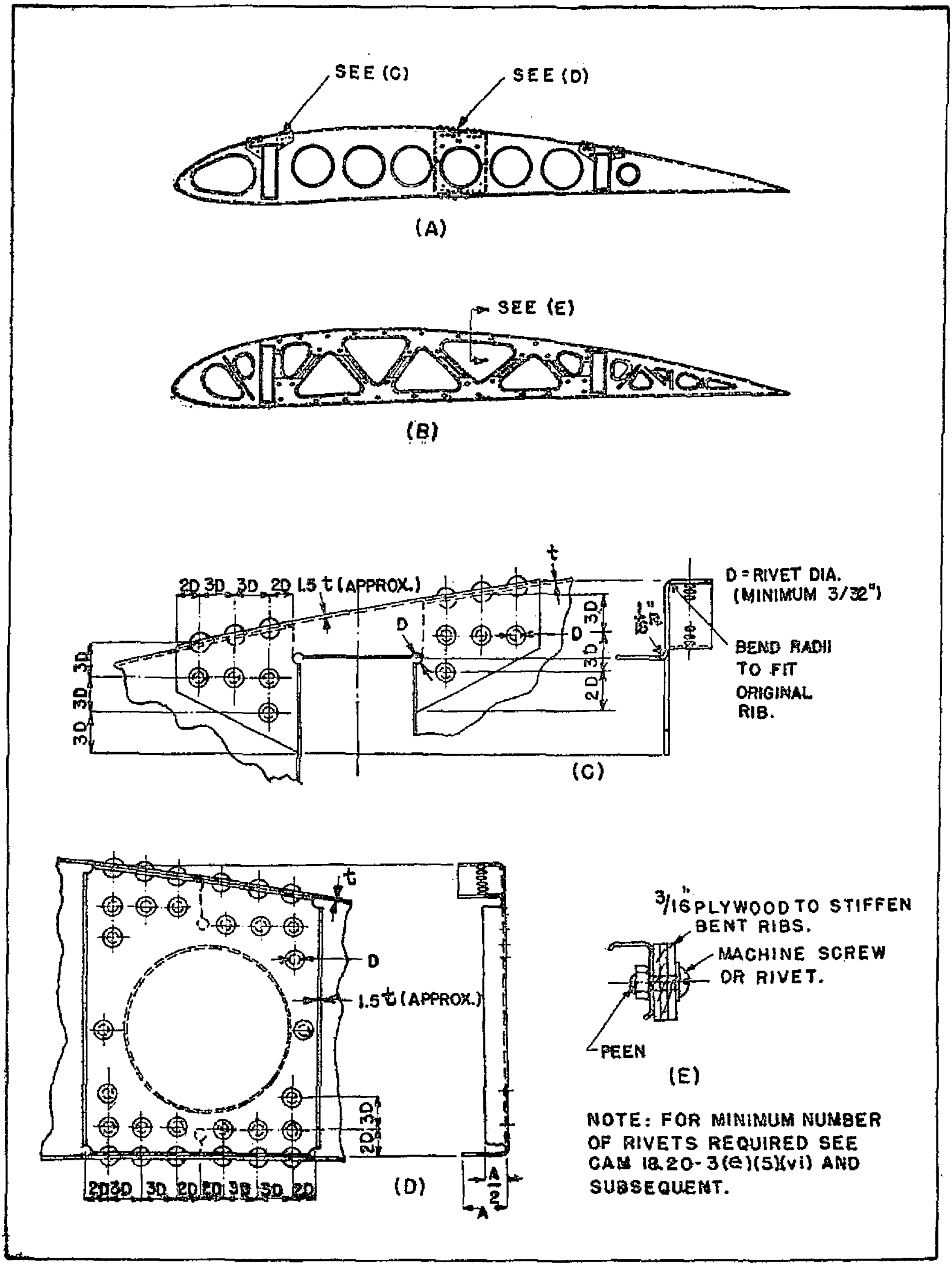


FIGURE 3-18. Typical Metal Rib Repairs (Usually Found on Small and Medium-Size Aircraft). Ref. *CAM 18.20-3(e)(5)(iv)*

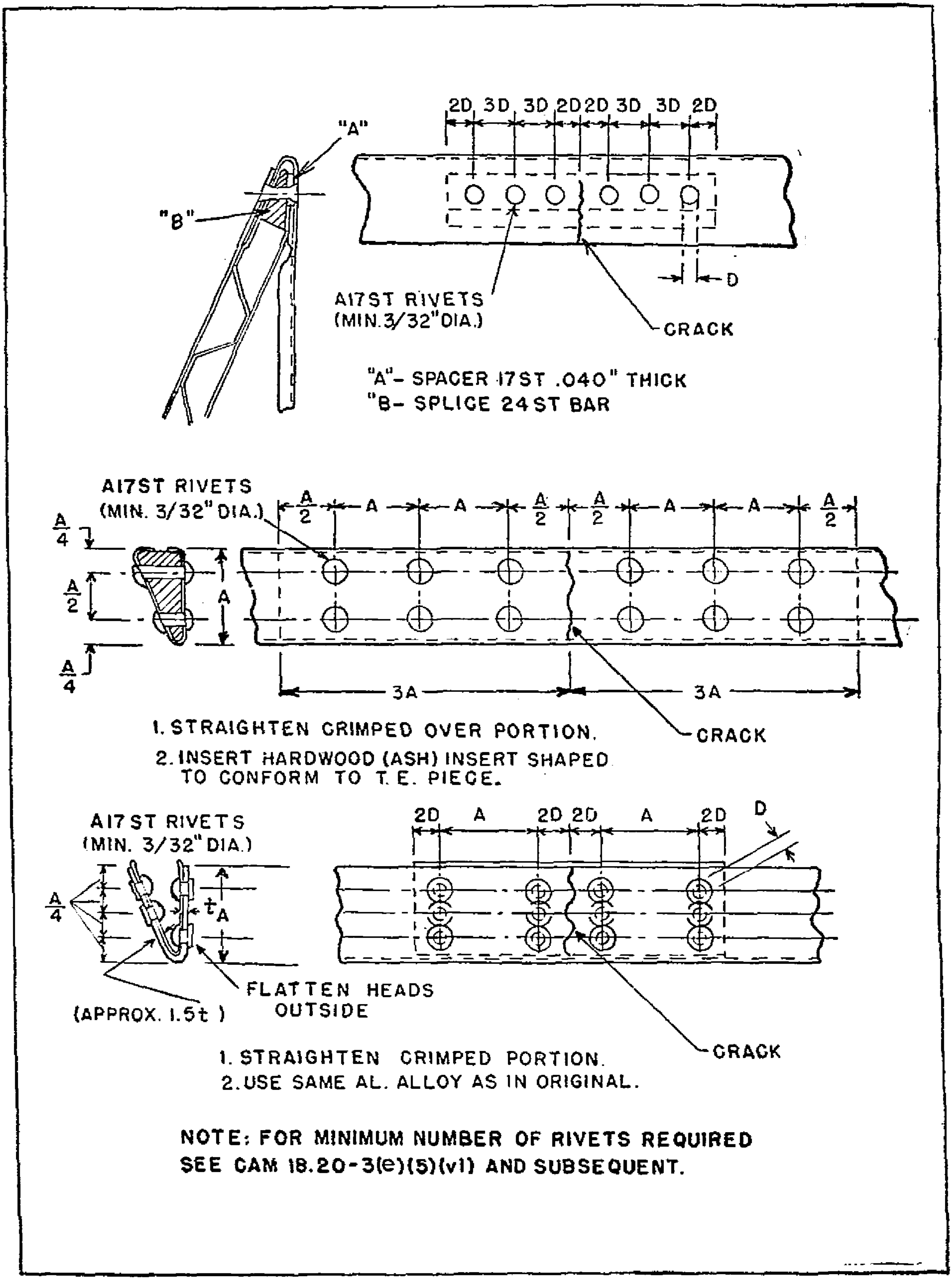
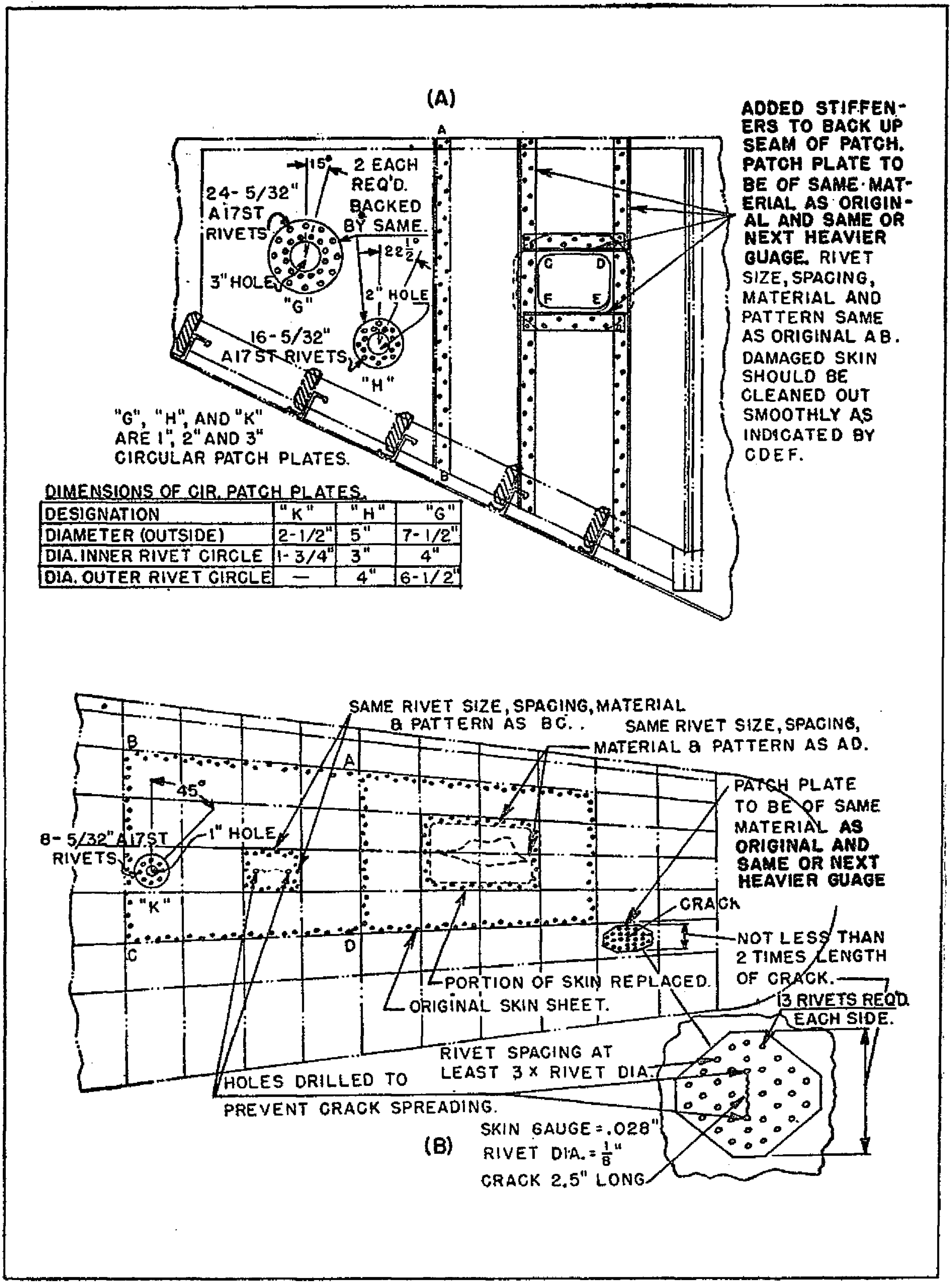


FIGURE 3-19. Typical Repairs of Trailing Edges. Ref. CAM 18.20-3(e)(5)(iv)(a)



DIMENSIONS OF CIR. PATCH PLATES.

DESIGNATION	"K"	"H"	"G"
DIAMETER (OUTSIDE)	2-1/2"	5"	7-1/2"
DIA. INNER RIVET CIRCLE	1-3/4"	3"	4"
DIA. OUTER RIVET CIRCLE	—	4"	6-1/2"

FIGURE 3-20. Typical Repairs of Stressed Sheet Covering. Ref. CAM 18.20-3(e)(5)(v)(*a*), 18.20-3(e)(5)(v)(*b*)

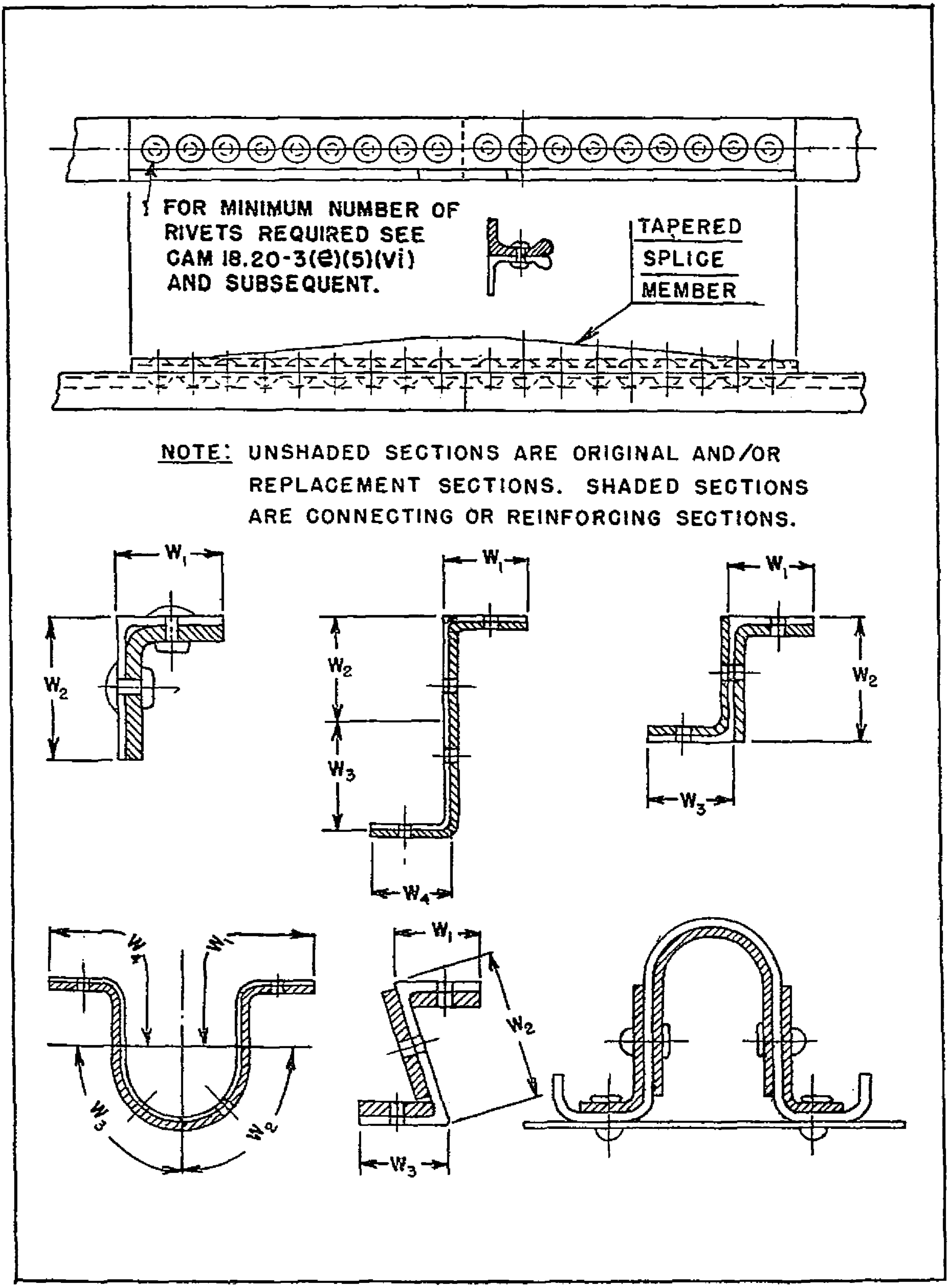


FIGURE 3-21. Typical Stringer and Flange Splices. Ref. CAM 18.20-3(e)(5)(viii), 18.20-3(e)(5)(viii)(b)

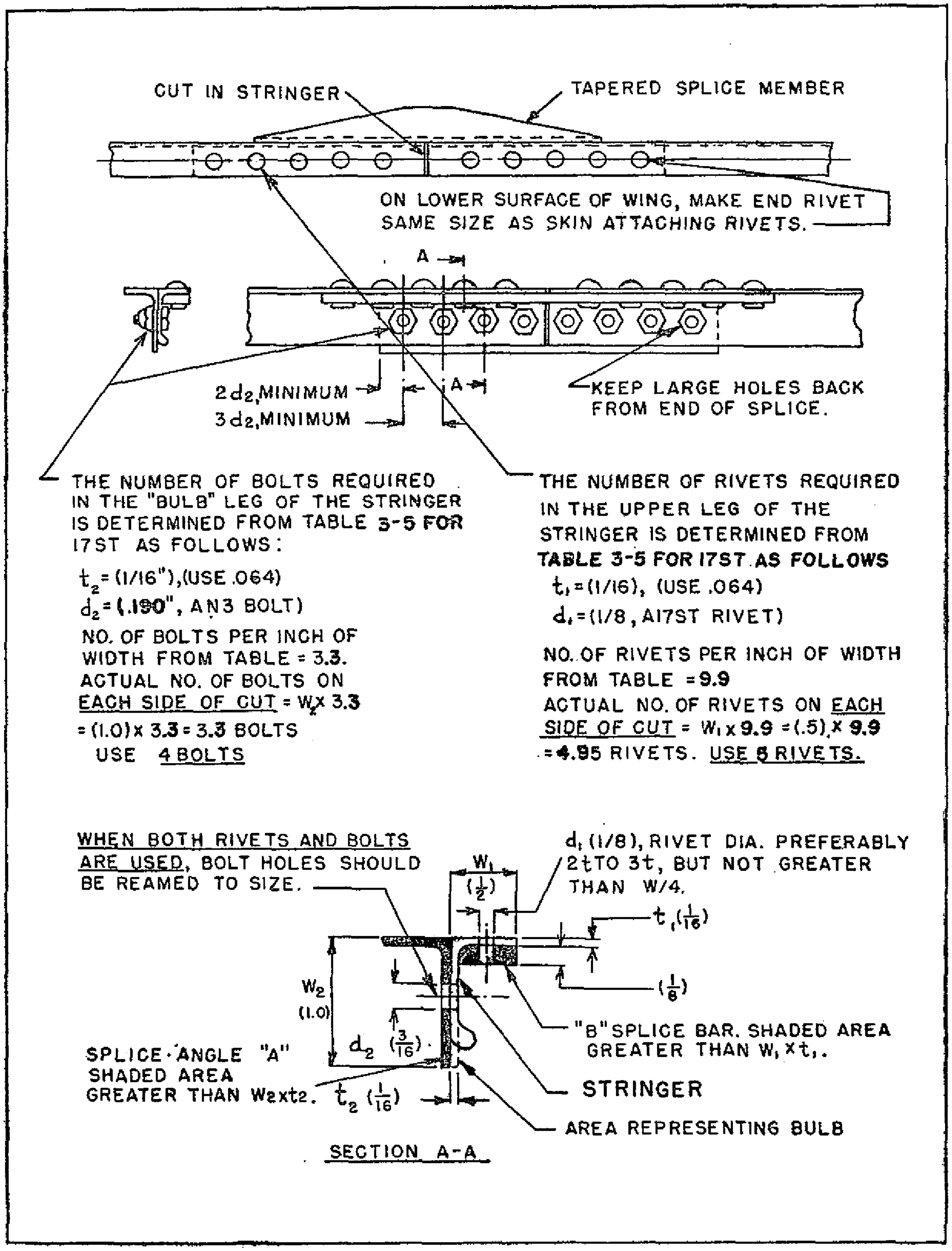
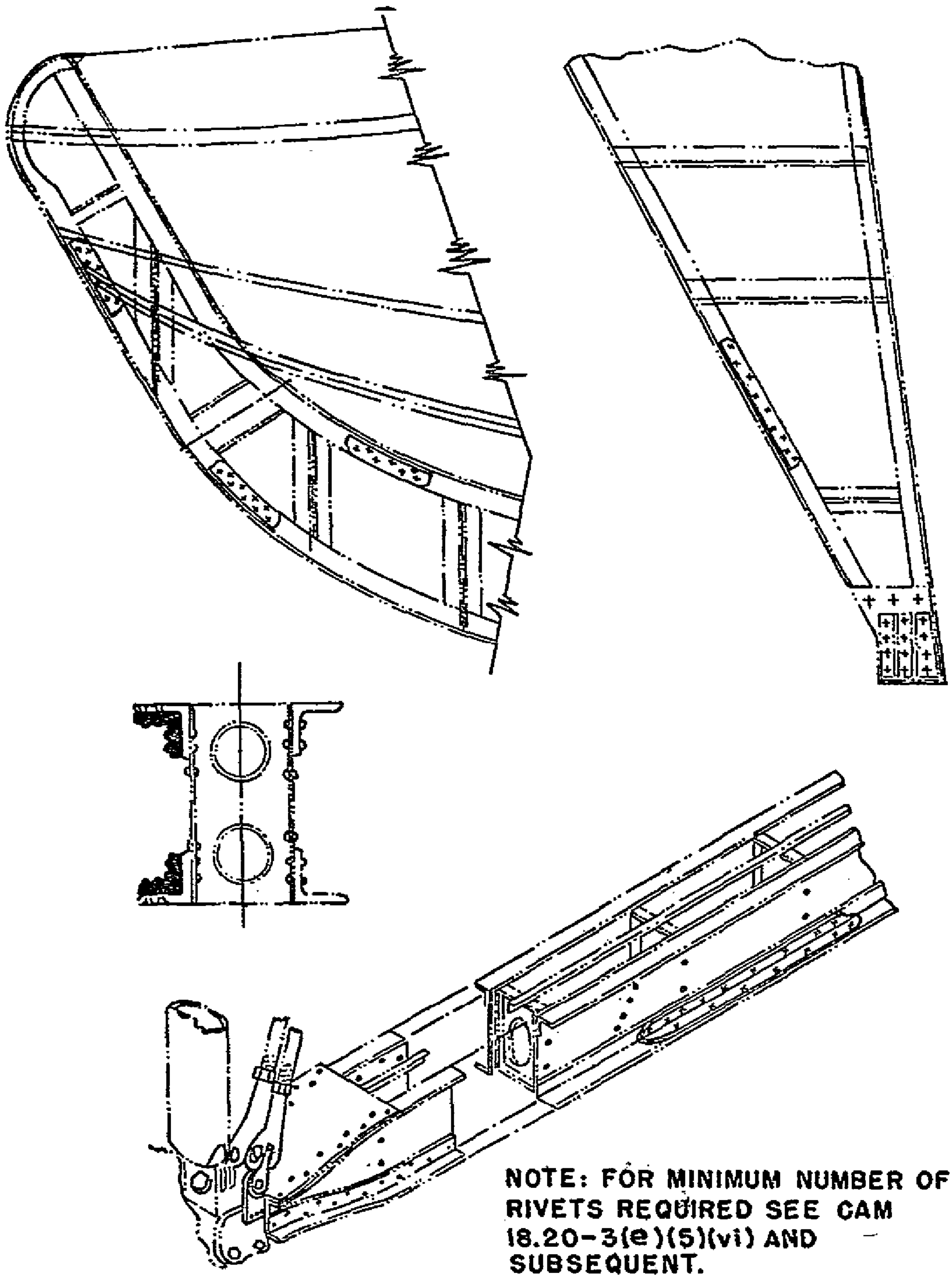


FIGURE 3-22. Example of Stringer Splice (Material—17ST AL Alloy). Ref. CAM 18.20-3(e)(5)(viii), 18.20-3(e)(5)(viii)(b), 18.20-3(e)(5)(viii)(c), 18.20-3(e)(5)(viii)(d)

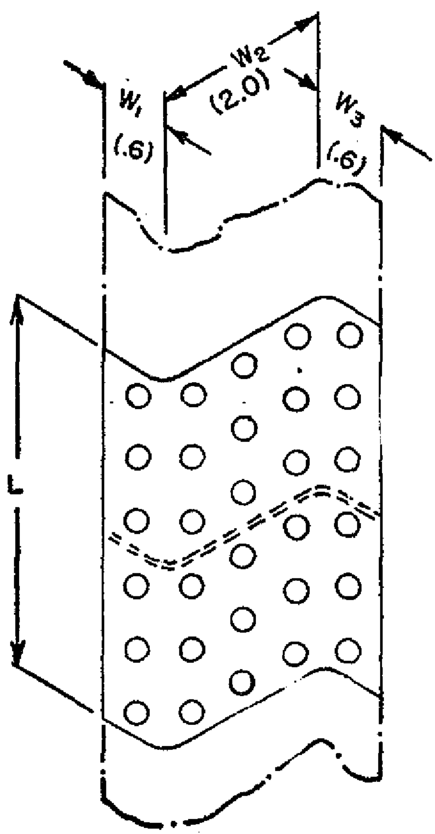


NOTE: STRENGTH INVESTIGATION USUALLY REQUIRED FOR THIS TYPE OF REPAIR.

FIGURE 3-23. Application of Typical Flange Splices and Reinforcement.
Ref. CAM 18.20-3(e)(5)(viii)

THE NUMBER OF RIVETS REQUIRED IN EACH LEG ON EACH SIDE OF THE CUT IS DETERMINED BY THE WIDTH "W", THICKNESS OF FRAME MATERIAL "t" AND RIVET DIAMETER "d", USING **TABLE 3-5 (FOR 17ST)** IN A MANNER SIMILAR TO THAT FOR STRINGERS, FIG. **3-22.**

NOTE (b), **TABLE 3-5 INDICATES** THAT ONLY 60% OF THE NUMBER OF RIVETS SO CALCULATED NEED BE USED IN SPLICES IN INTERMEDIATE FRAMES.



EXAMPLE:

FLANGE LEG

t = (.040)

d_1 = (1/8, A17ST RIVET)

W_1 = (.6) IN.

NO. OF RIVETS PER IN. OF WIDTH, **FROM TABLE 3-5 (FOR 17ST) = 6.2**

NO. OF RIVETS IN LEG = W_1 x **6.2** = (.6) x **6.2** = 3.72, SAY 4 RIVETS.
60% OF 4 = .6 x 4 = 2.4 RIVETS.

USE 3 RIVETS, EACH SIDE OF CUT.

BACK OF ZEE (OR CHANNEL)

t = (.040)

d_2 = (1/8, A17ST RIVET)

W_2 = 2.0 INS.

NO. OF RIVETS PER IN. OF WIDTH, **FROM TABLE 3-5 = 6.2 = (2) X 6.2 = 12.4**
SAY 13 RIVETS. 60% OF 13 = 7.8 RIVETS.

USE 8 RIVETS, EACH SIDE OF CUT.

"L" SHOULD BE MORE THAN TWICE W_2

THICKNESS OF SPLICE PLATE GREATER THAN THAT OF FRAME

FIGURE 3-24. Example of Splice and Intermediate Frame (Material—17ST AL Alloy). Ref. CAM 18.20-3(e) (5) (viii) (e)

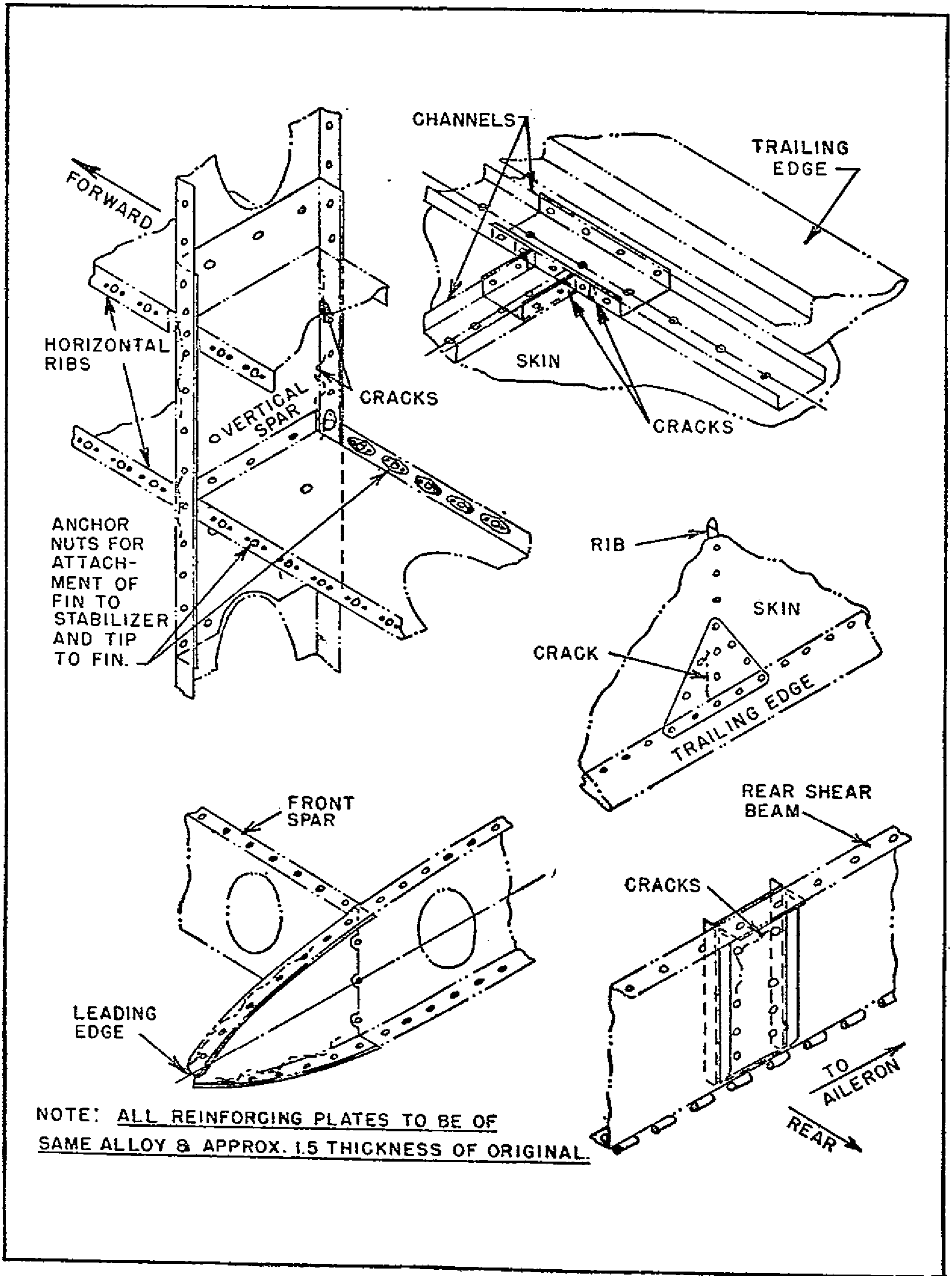


FIGURE 3-25. Typical Methods of Repairing Cracked Leading and Trailing Edges and Rib Intersections. Ref. CAM 18.20-3(e)(5)(ix)

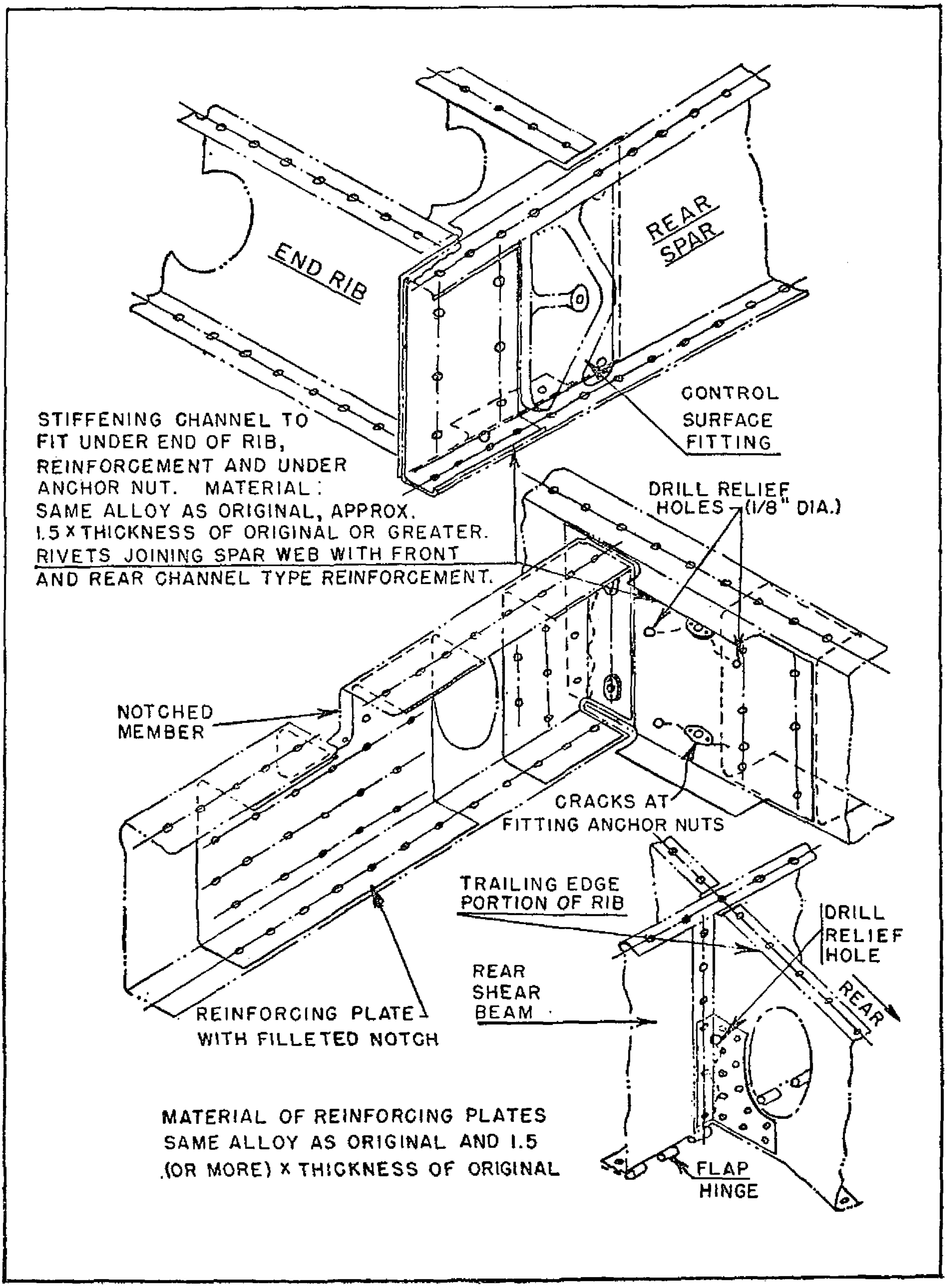


FIGURE 3-26. Typical Methods of Replacing Cracked Members at Fittings. Ref. CAM 18.20-3(e)(5)(ix)

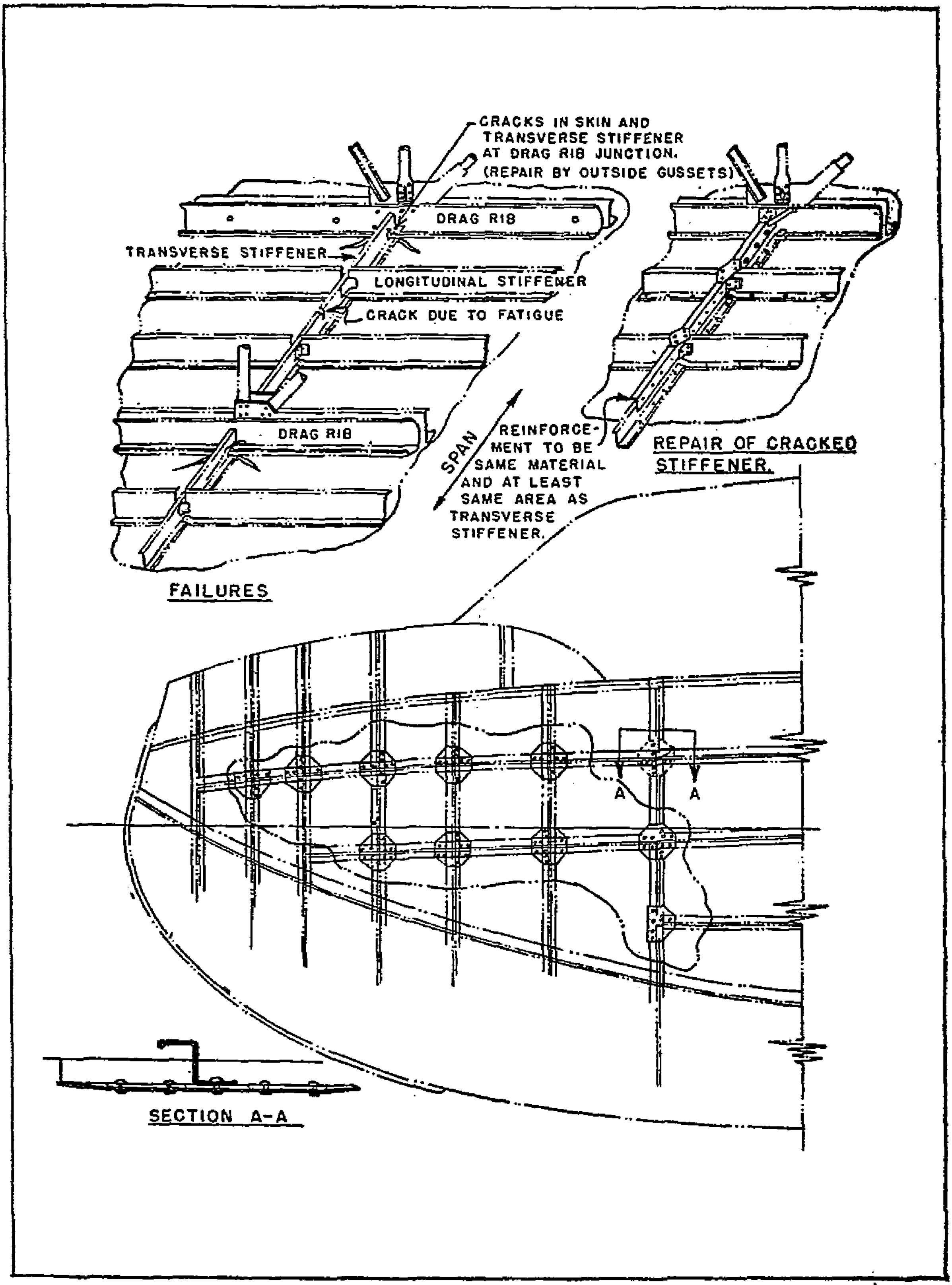


FIGURE 3-27. Typical Methods of Repairing Cracked Frame and Stiffener Combinations. Ref. CAM 18.20-3(e) (5) (ix)

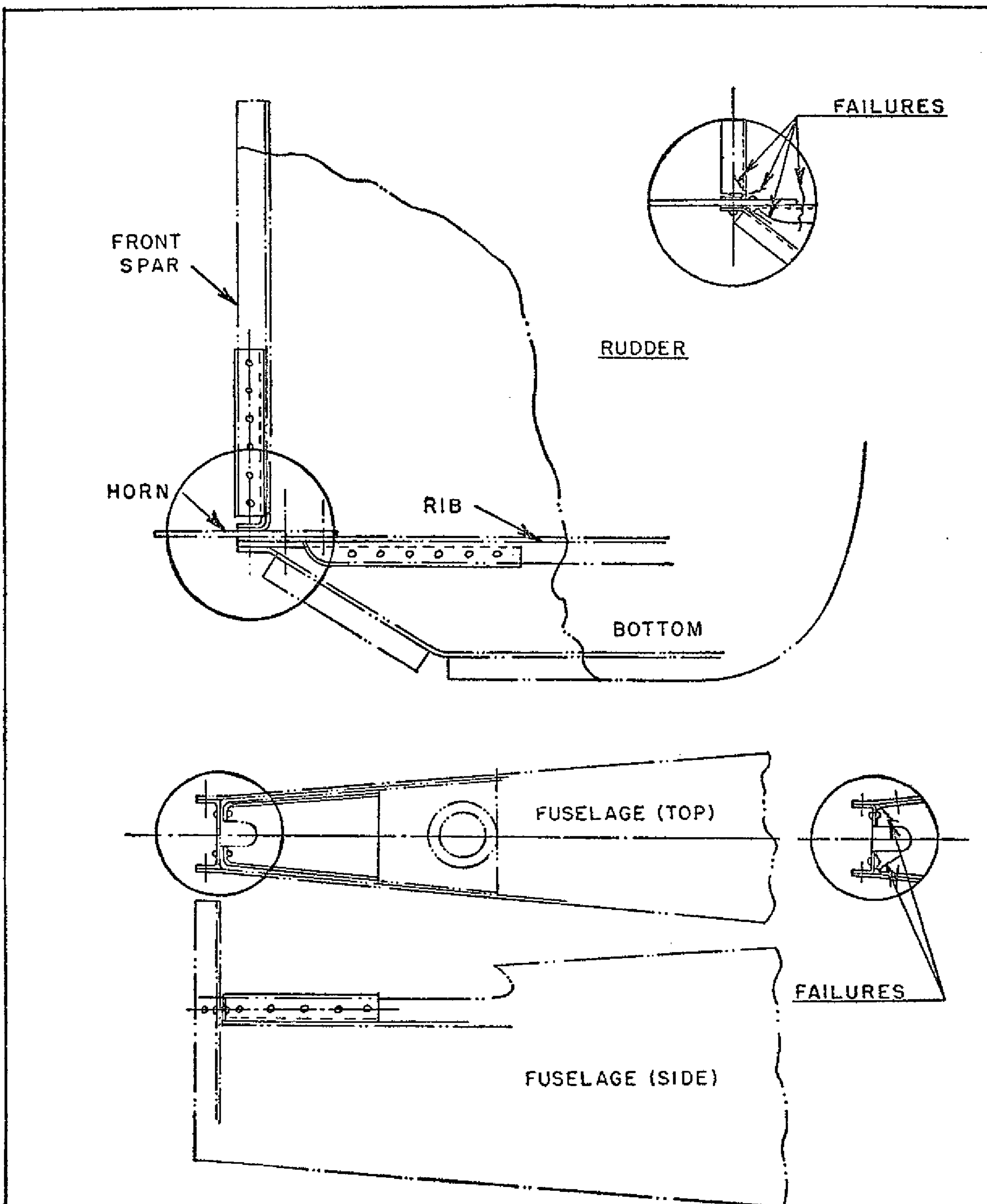


FIGURE 3-28. Typical Repairs to Rudder and to Fuselage at Tail Post.
Ref. CAM 18.20-3(e) (5) (ix)

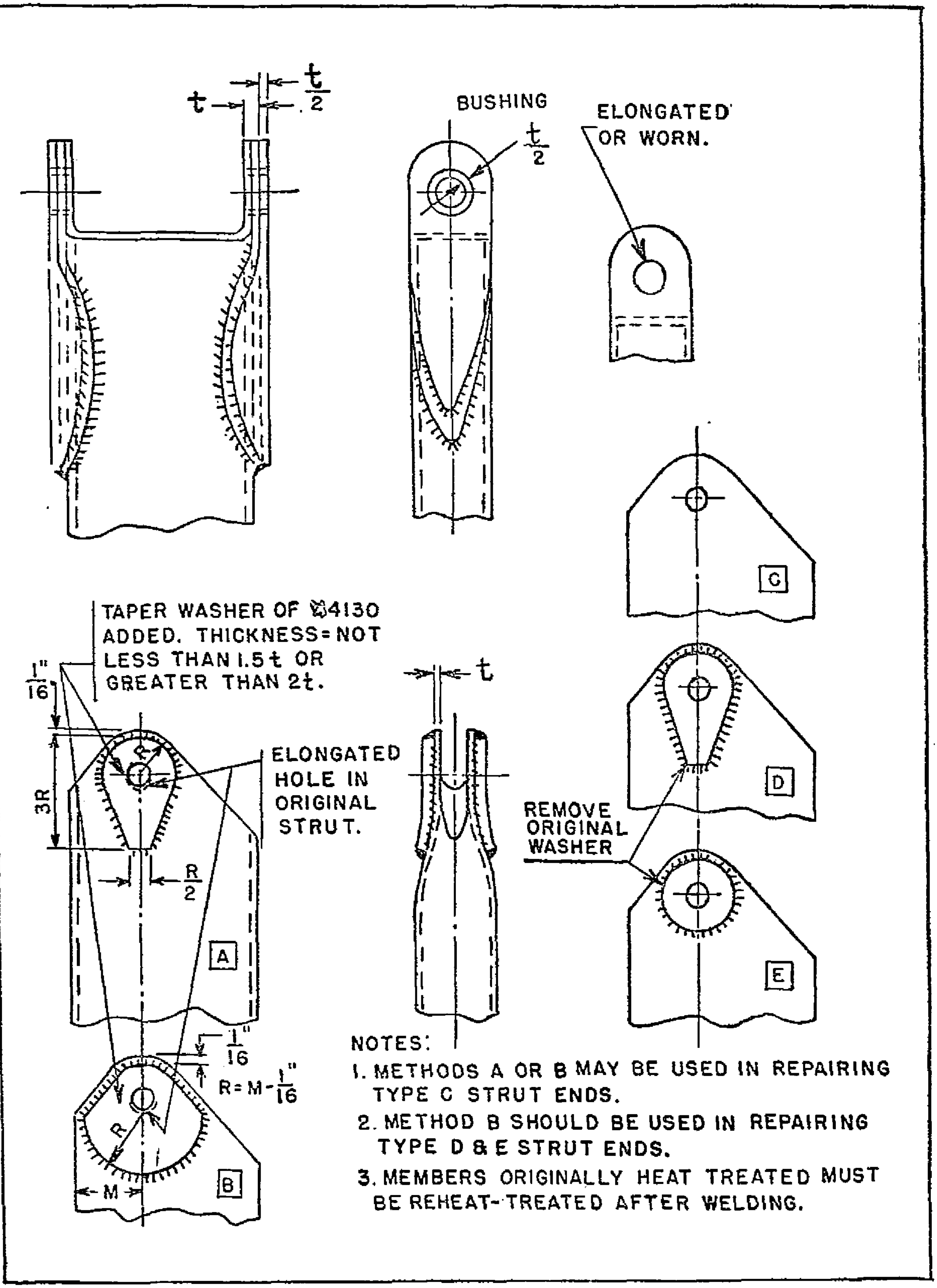


FIGURE 3-29. Typical Methods of Repairing Elongated or Worn Bolt Holes.
Ref. CAM 18.20-3(e)(6)(i)(c)

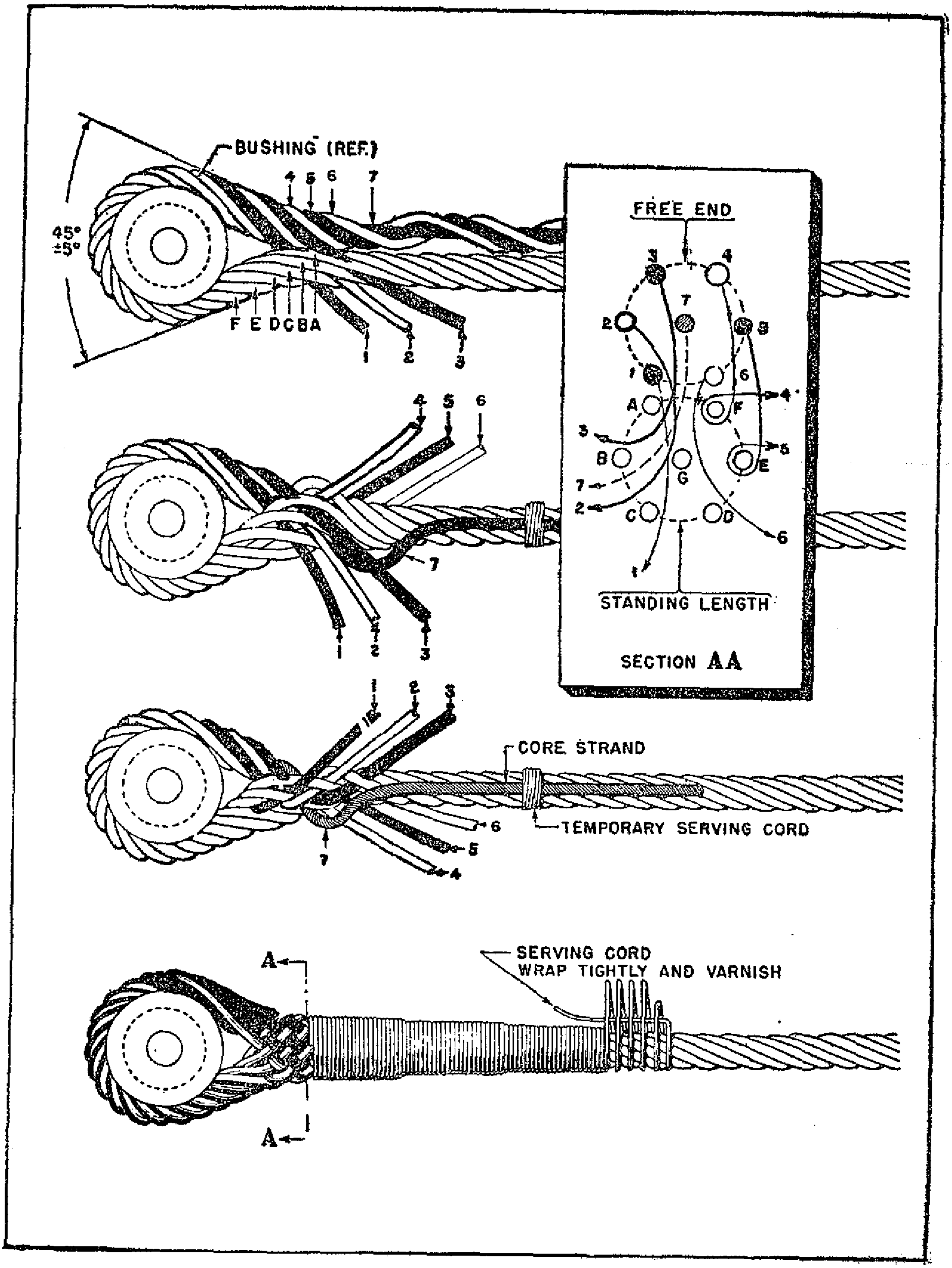
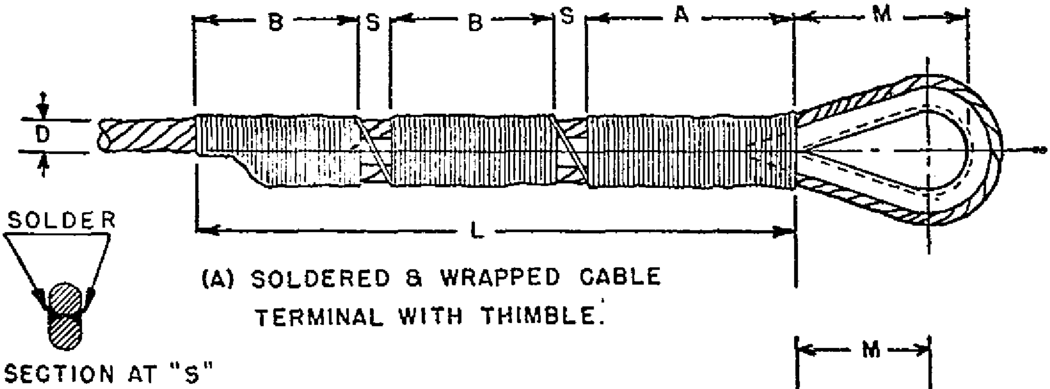
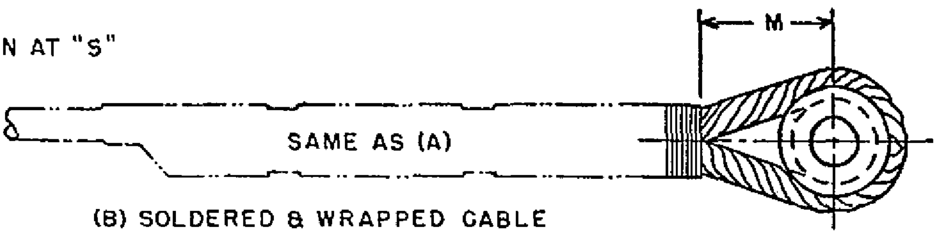


FIGURE 4-1. Preparation of a Woven Cable Splice. Ref. CAM 18.20-4(c)



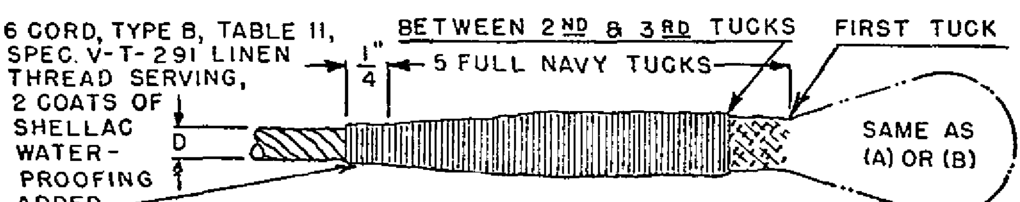
(A) SOLDERED & WRAPPED CABLE TERMINAL WITH THIMBLE.



(B) SOLDERED & WRAPPED CABLE TERMINAL WITH BUSHING.

D	L	A	B	M	S	WRAPPING WIRE #48-19*		SPECIFICATION NO.	
		PLUS OR MINUS 1/32"				DIA. INCH.	APPROX. LENGTH	THIMBLE (A)	BUSHING (B)
3/32	2-1/4	3/4	5/8	3/4	1/8	.020	37"	AN-100-3	AN-111-3
1/8	2-3/4	1	3/4	3/4	1/8	.025	58"	AN-100-4	AN-111-4
5/32	3-3/8	1-1/8	1	7/8	1/8	.025	82"	AN-100-5	AN-111-5
3/16	3-5/8	1-1/4	1	1-1/8	3/16	.035	109"	AN-100-6	AN-111-6
7/32	4	1-3/8	1-1/8	1-1/4	3/16	.035	—	AN-100-7	AN-111-7
1/4	4-1/2	1-1/2	1-1/4	1-1/2	1/4	.035	159"	AN-100-8	AN-111-8
5/16	5-1/4	1-3/4	1-1/2	1-7/8	1/4	.050	195"	AN-100-10	AN-111-10
3/8	6-1/4	2-1/4	1-3/4	2-1/8	1/4	.050	—	AN-100-12	AN-111-12
7/16	7	2-1/2	2	2-1/2	1/4	.050	—	AN-100-14	—
1/2	8	2-3/4	2-1/4	2-7/8	3/8	.050	—	AN-100-16	—

*ARMY SPECIFICATION- AN-QQ-W-435



(C) SPLICED CABLE TERMINAL

FIGURE 4-2. Wrapped or Spliced Cable Terminals. Ref. CAM 18.20-4(d)

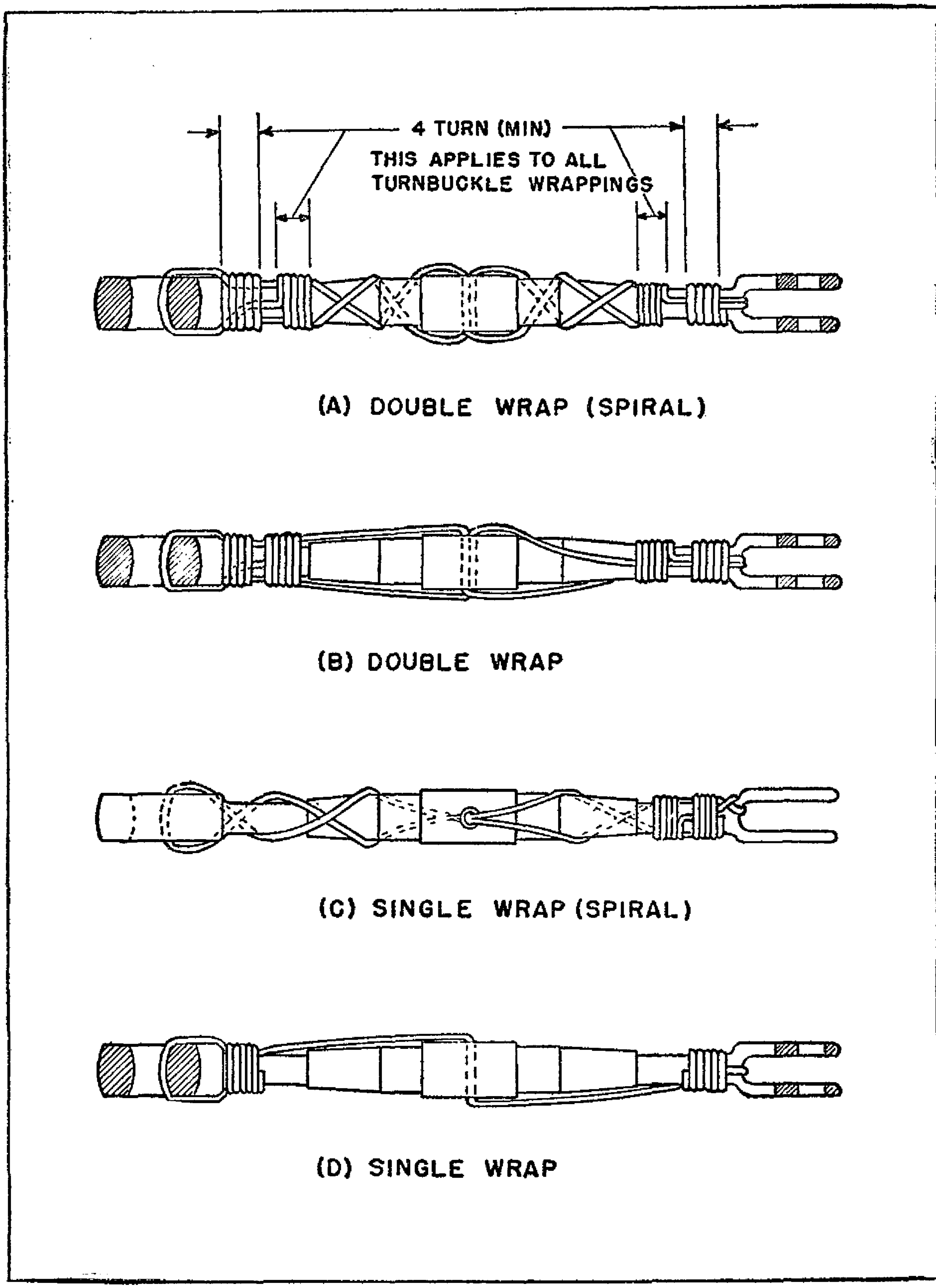


FIGURE 4-3. Safetying Turnbuckles. Ref. CAM 18.20-4(e)(1)

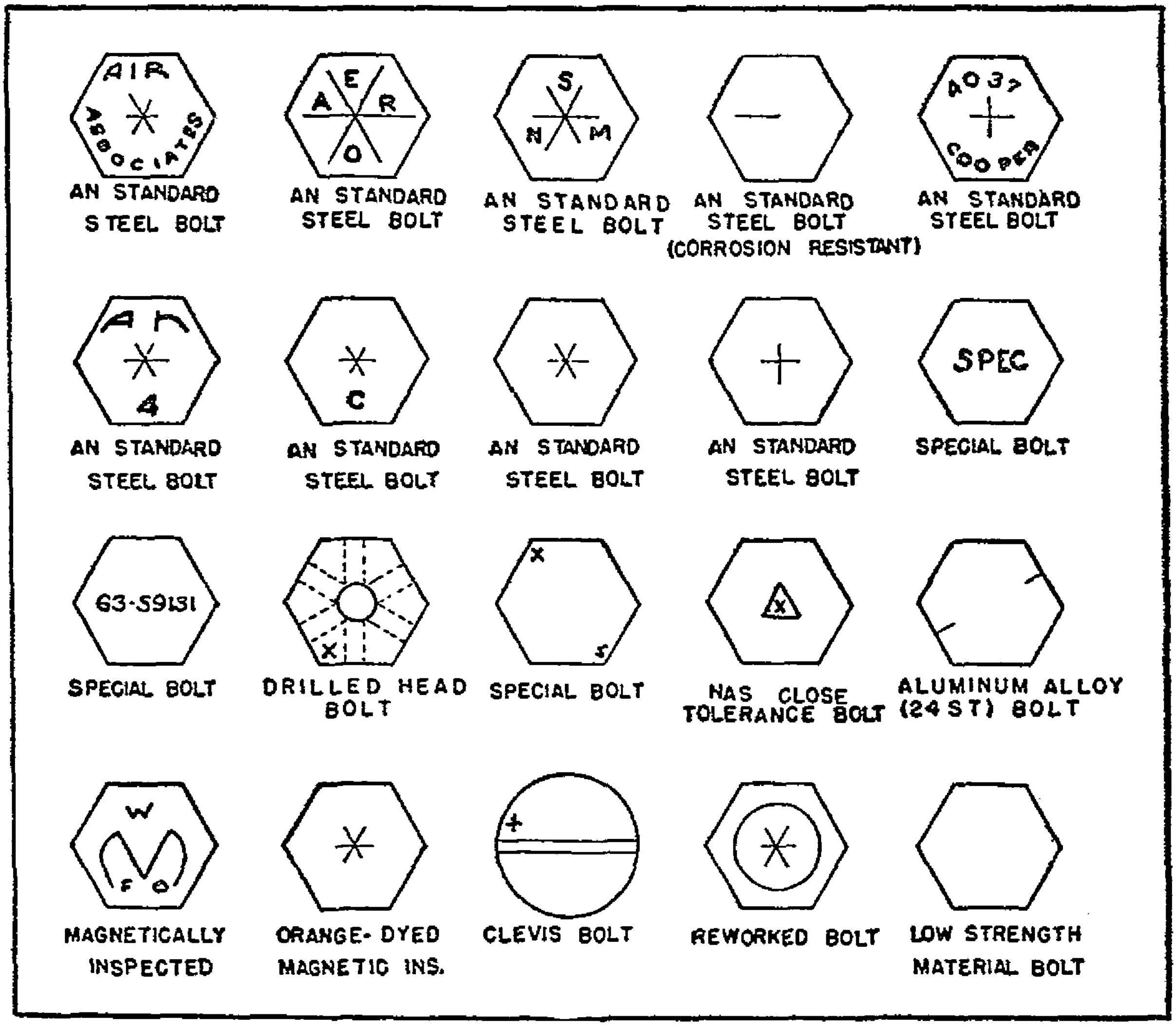


FIGURE 5-1. Bolt Identification. Ref. CAM 18.20-5(a)(1)

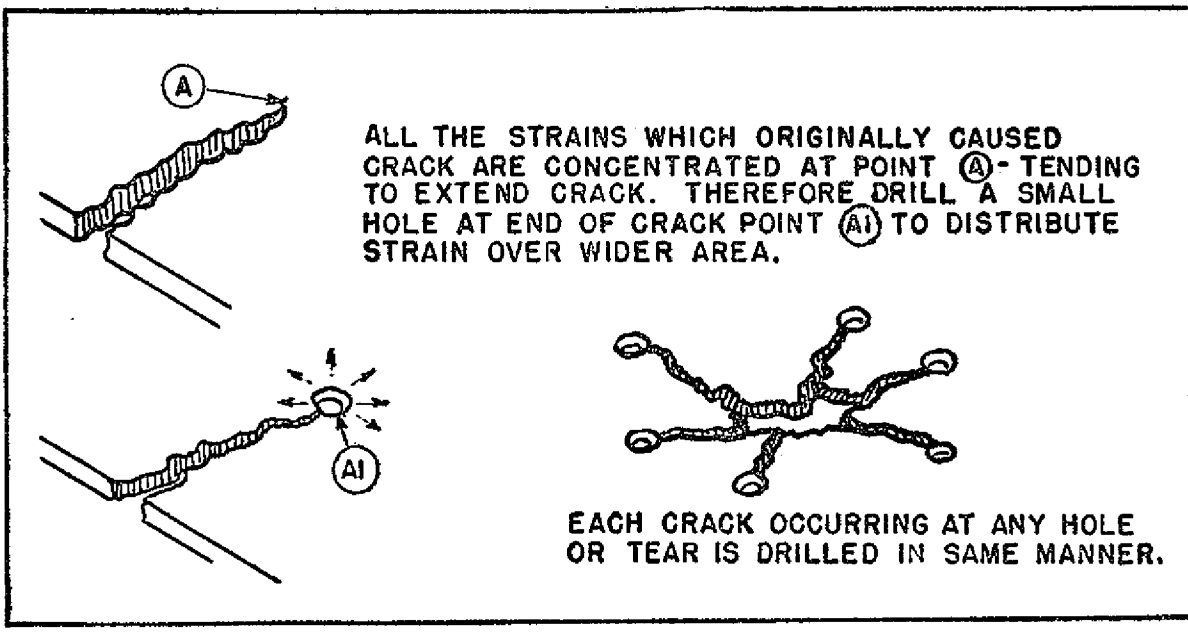


FIGURE 9-1. Stop-Drilling Cracks. Ref. CAM 18.20-9(e)

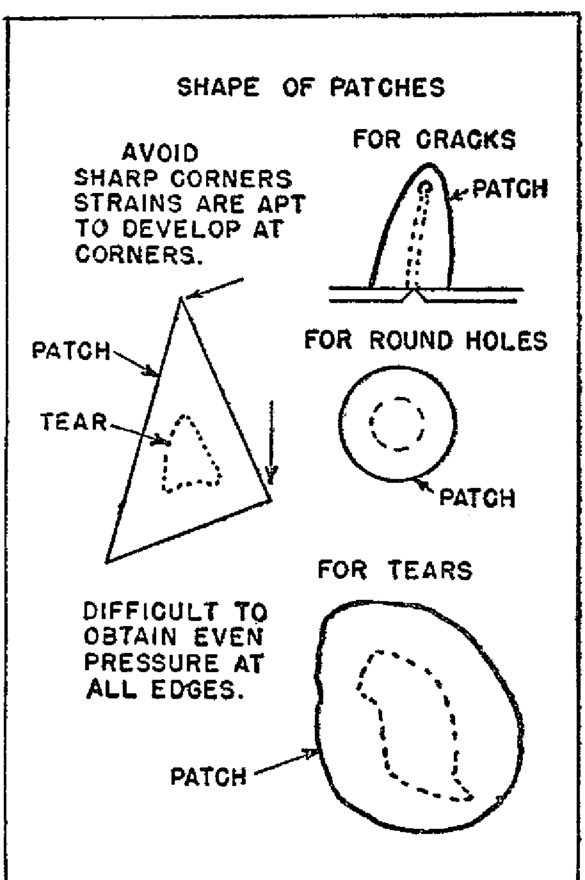


FIGURE 9-2. Surface Patches. Ref. CAM 18.20-9(e) (1)

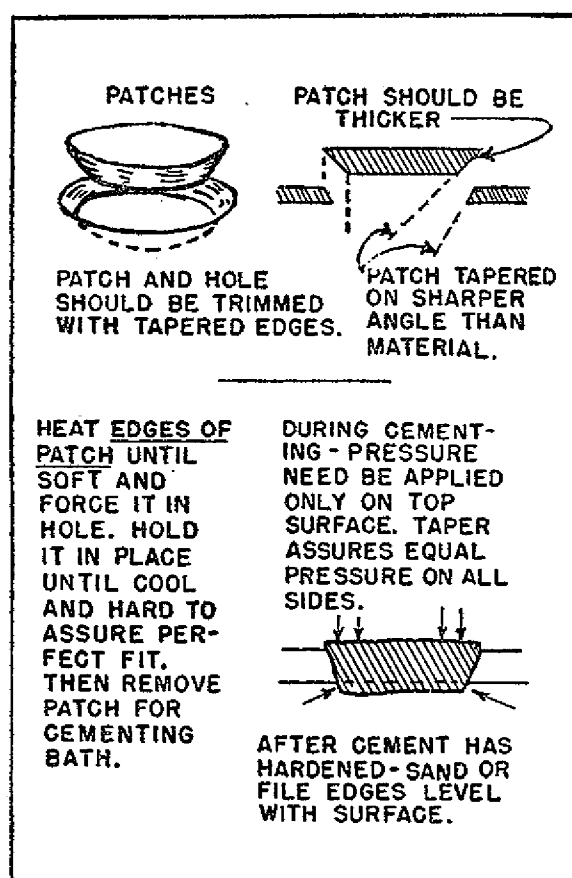


FIGURE 9-3. Plug-Patch Repair. Ref. CAM 18.20-9(e) (2)

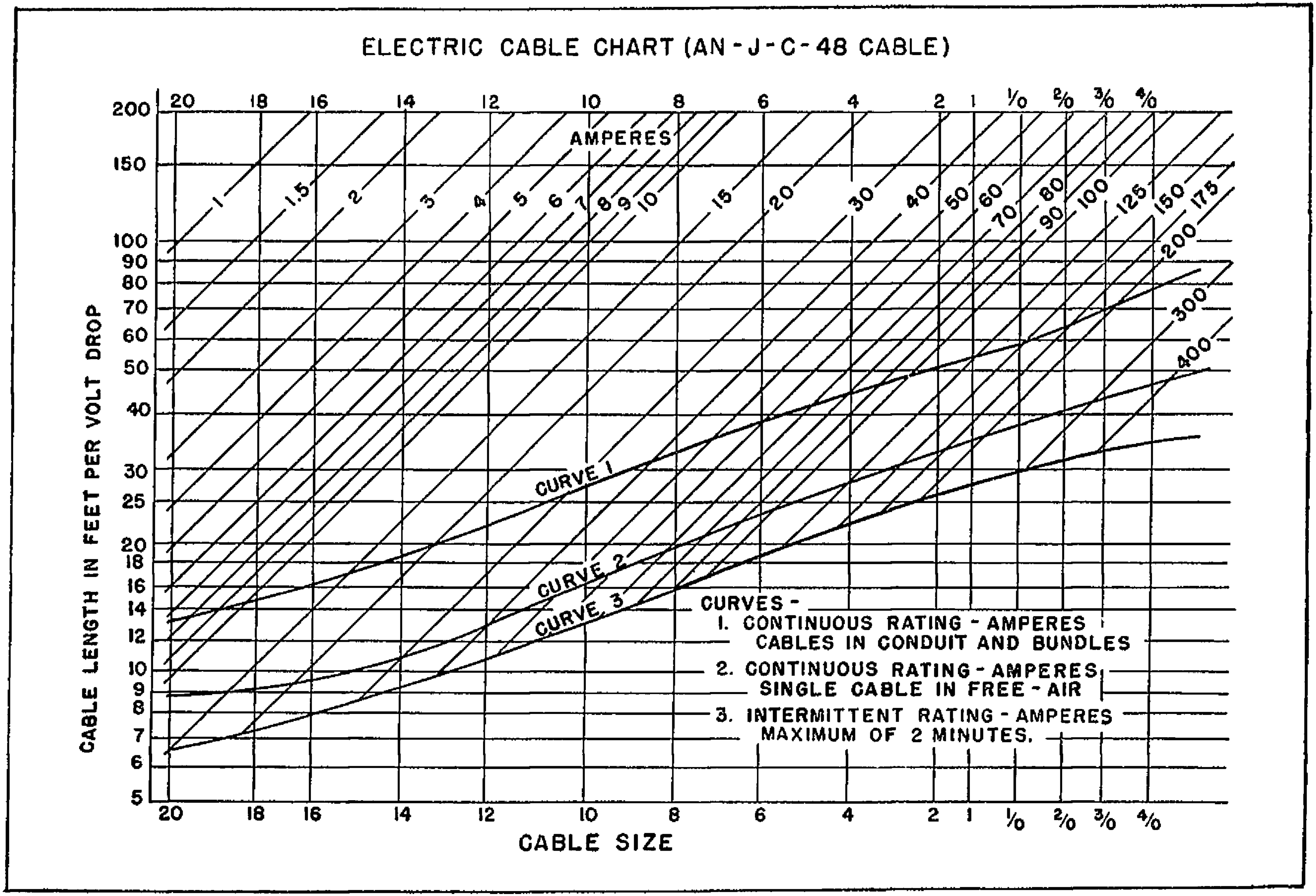


FIGURE 11–1. Electric Cable Chart (AN–S–C–48 Cable). Ref. CAM 18.20–11(e)(2)(ii)(*a*), 18.20–11(e)(2)(iii), 18.20–11(e)(2)(v)

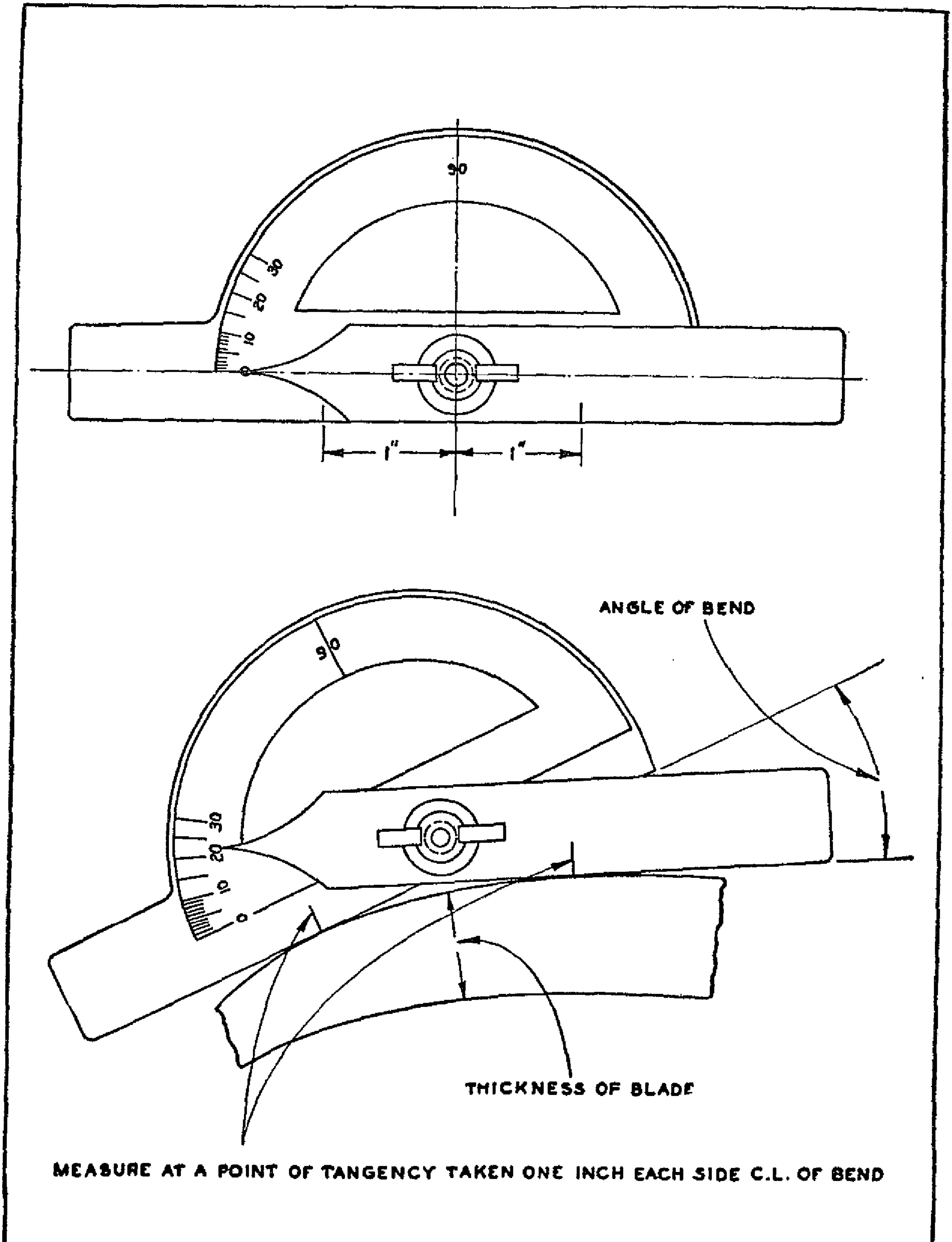


FIGURE 14-1. Protractor and Method of Measuring Angle of Bend in Aluminum-Alloy Propellers. Ref. CAM 18.20-14(e)(3)(iii)

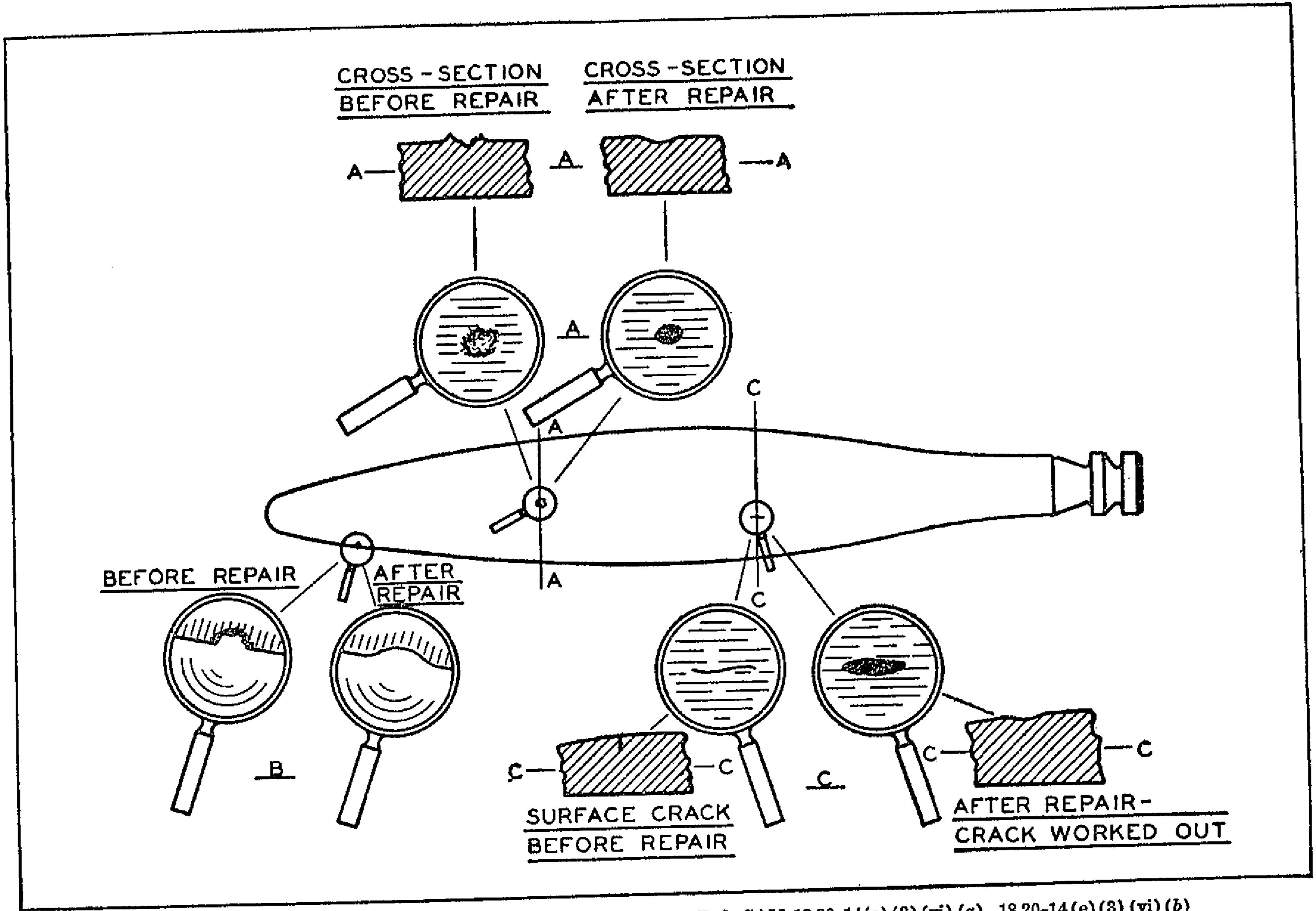


FIGURE 14-2. Method of Repairing Surface Cracks, Nicks, etc. Ref. CAM 18.20-14(e)(3)(vi)(*a*), 18.20-14(e)(3)(vi)(*b*)

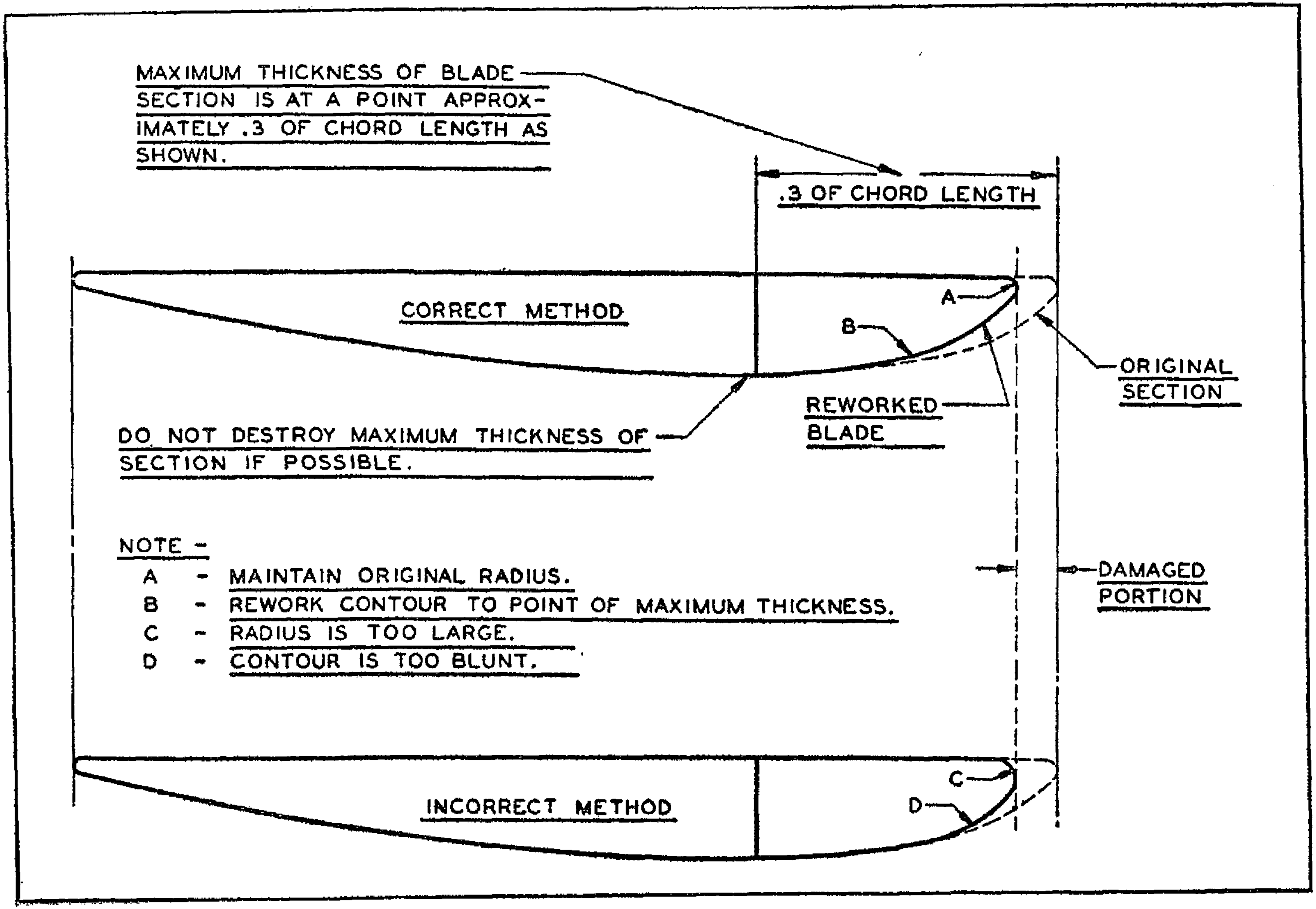


FIGURE 14-3. Correct and Incorrect Method of Reworking Leading Edge. Ref. CAM 18.20-14(e) (3) (vi) (b)

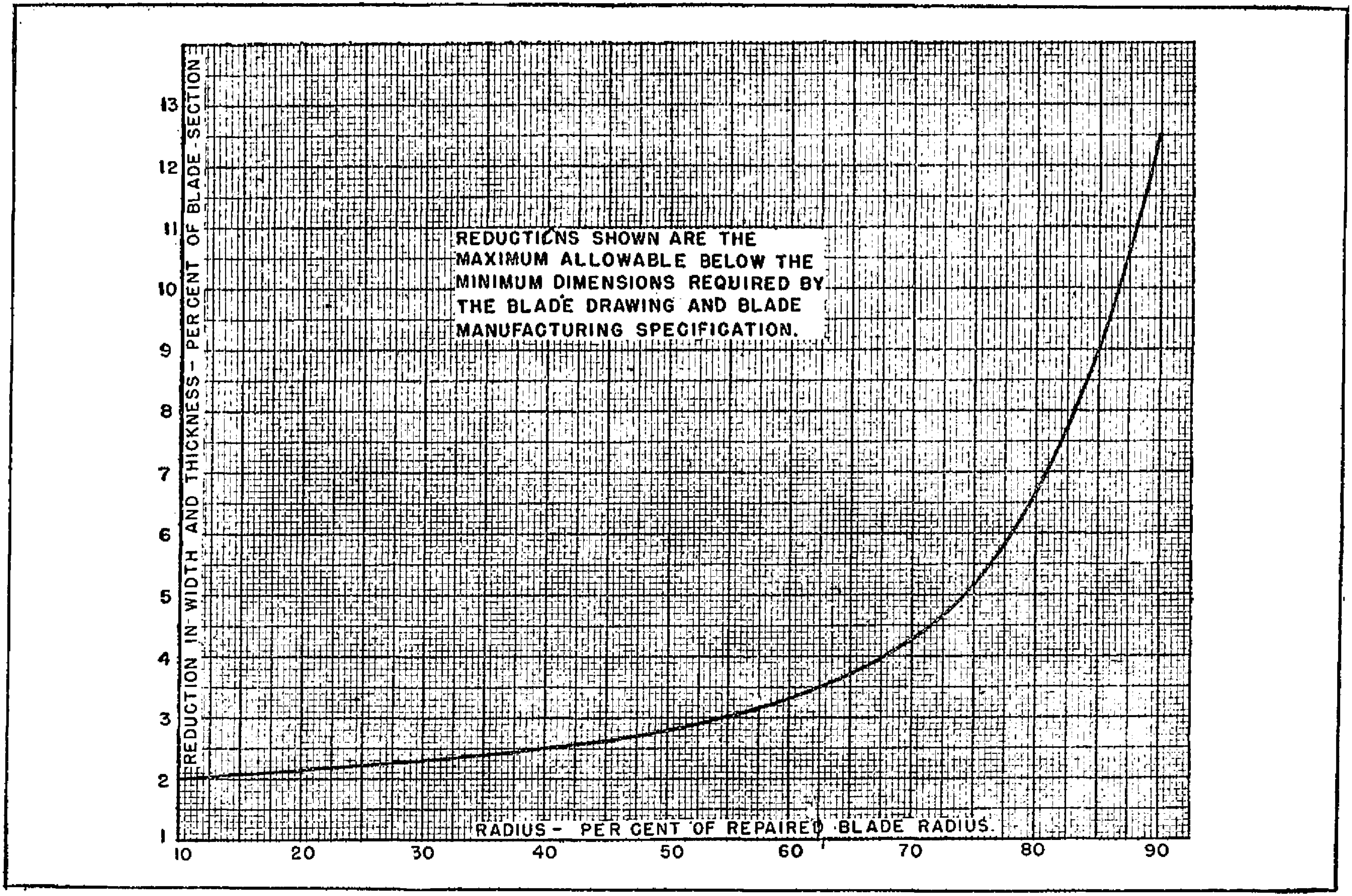


FIGURE 14-4. Repair Limits to Section Width and Thickness for Aluminum-Alloy Propeller Blades. Ref. CAM 18.20-14(e)(3)(iv)

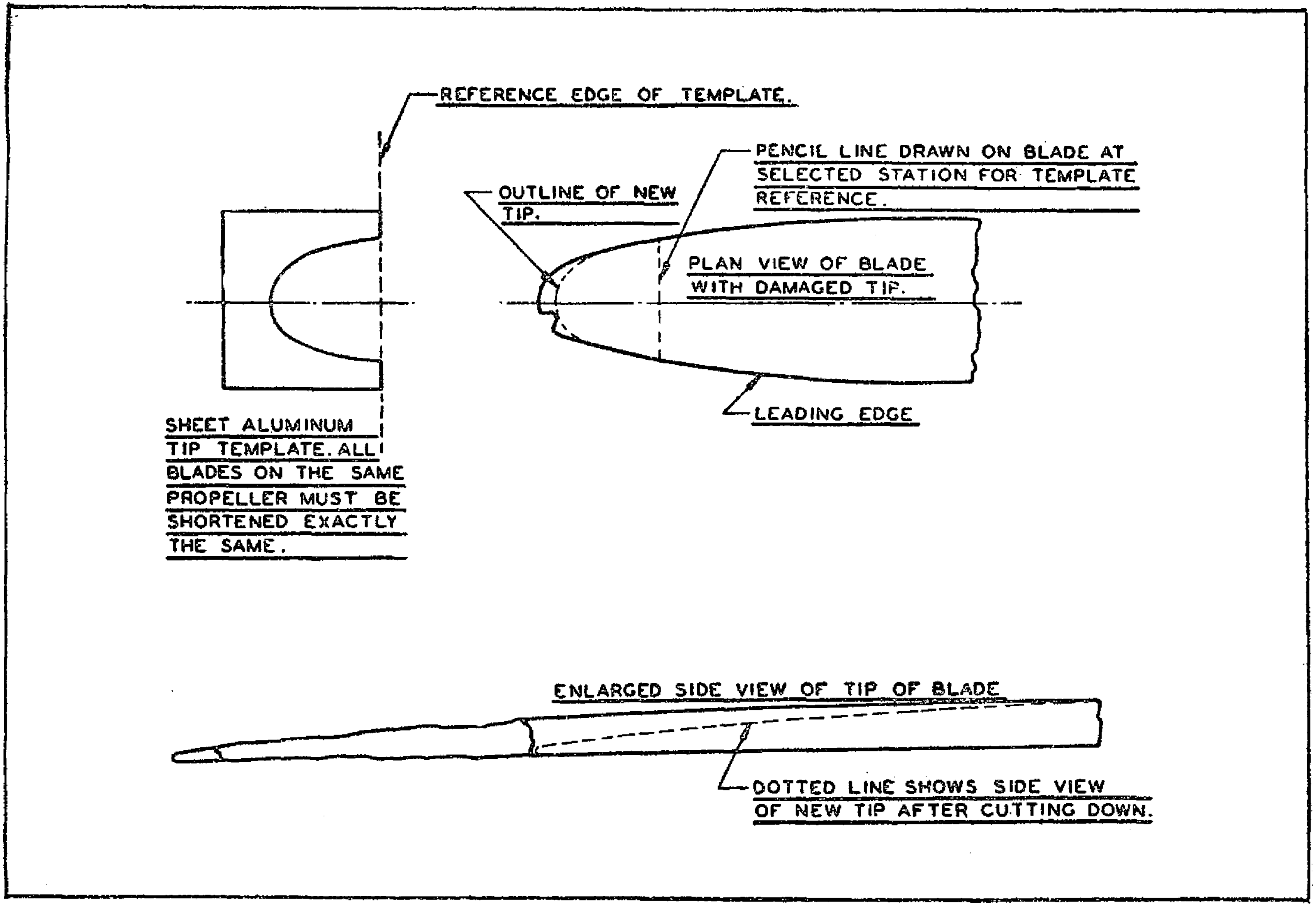


FIGURE 14-5. Method of Repairing Damaged Tip. Ref. CAM 18.20-14(e)(3)(ii)

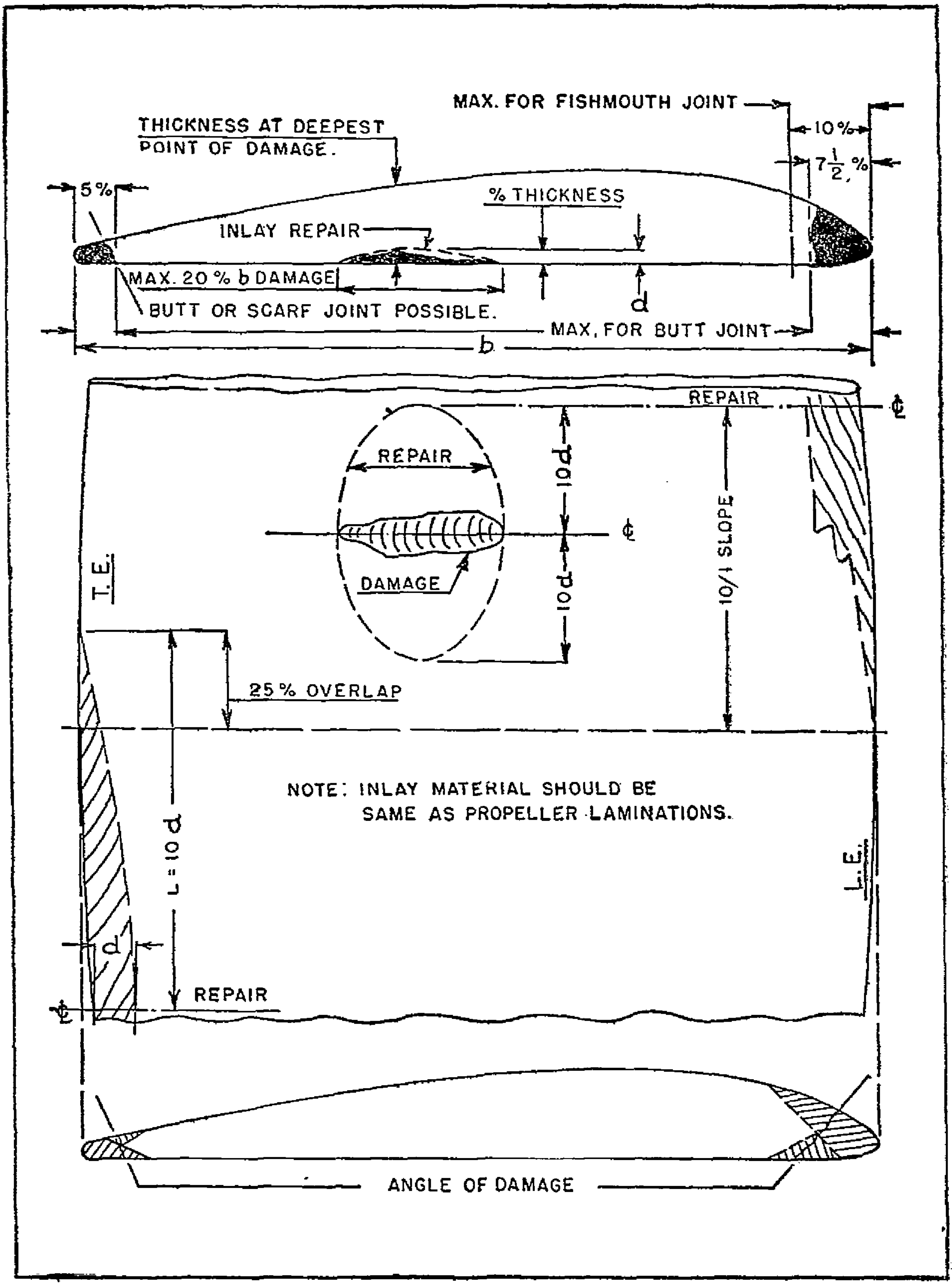


FIGURE 14–6. Propeller Repair by Addition of Small Inlay. Ref. CAM 18.20–14(f)(3)(iv)

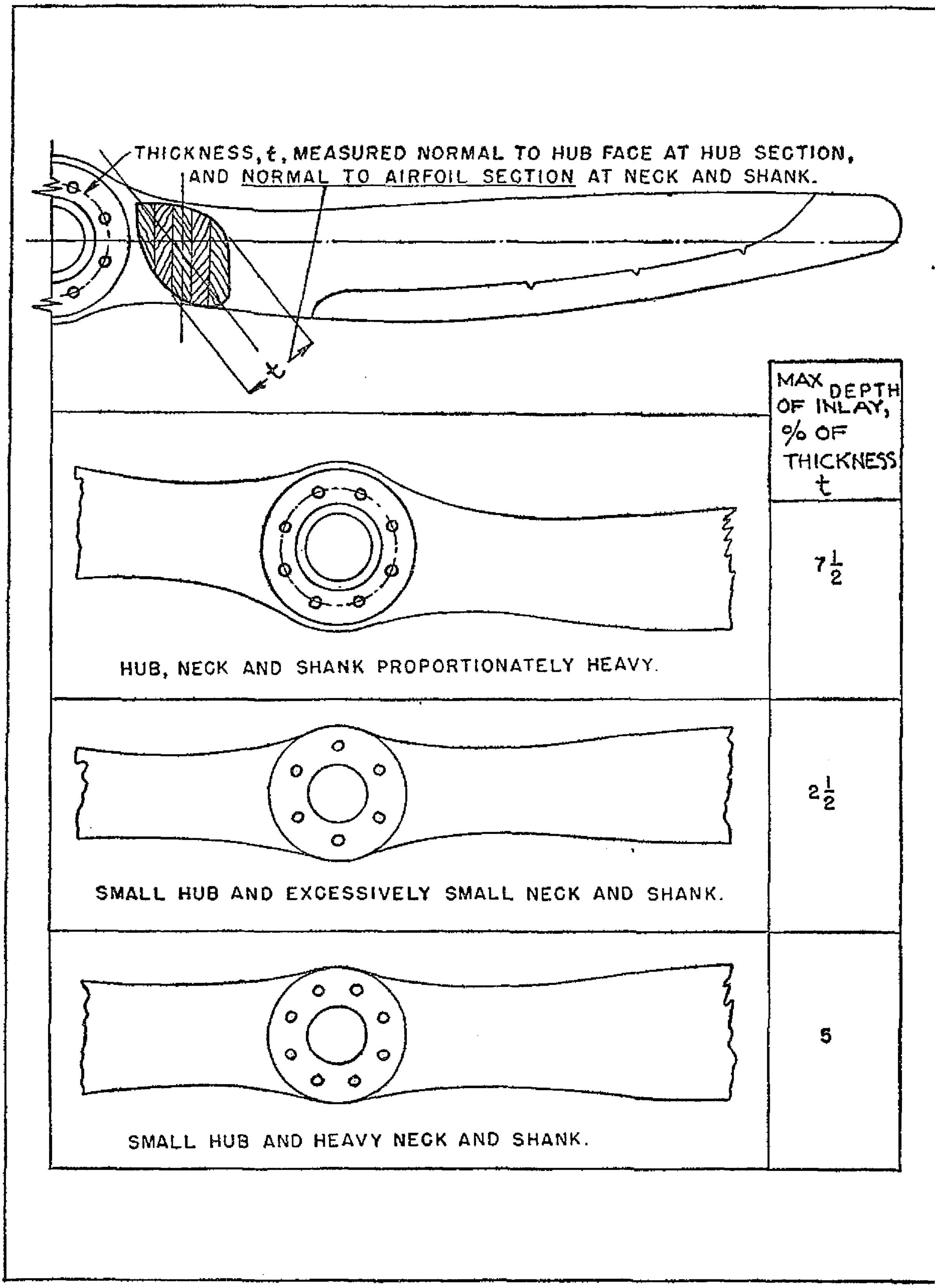
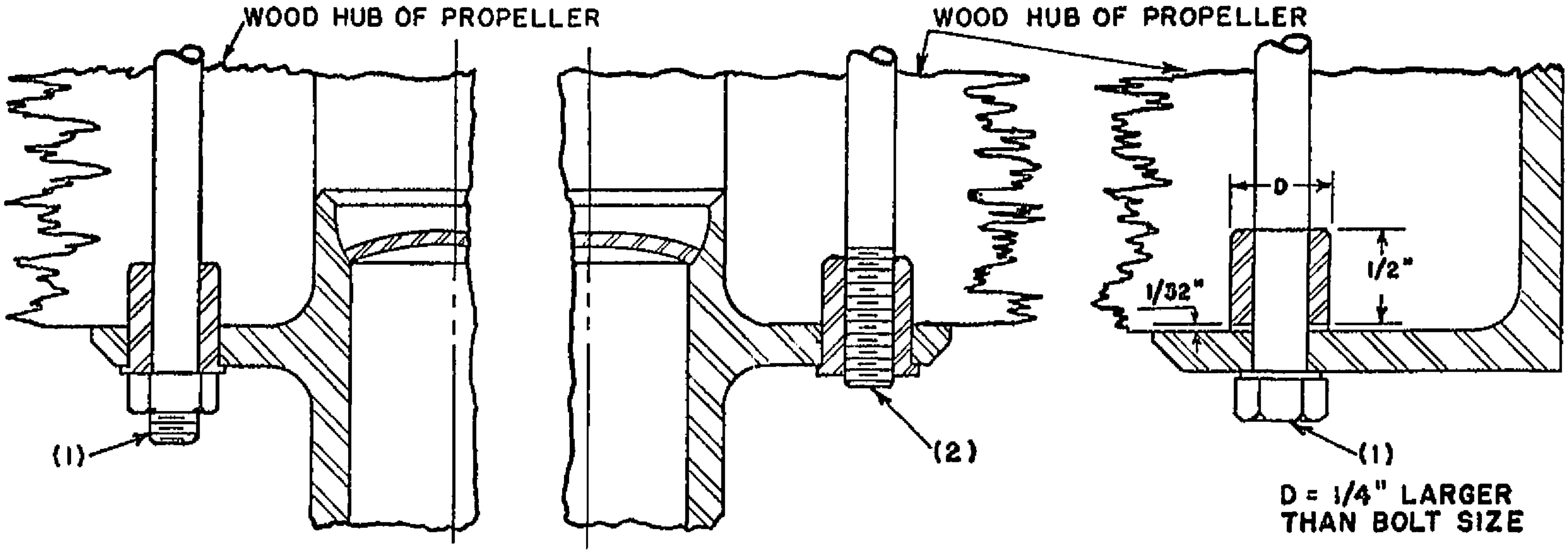


FIGURE 14-7. Splicing Propeller Laminations. Ref. CAM 18.20-14(f) (4)



METHOD (A) METHOD (B)

REPAIR OF DAMAGED OR ELONGATED BOLT HOLES IN PROPELLER HUB FLANGES

METHOD (C)

REPAIR OF ELONGATED BOLT HOLES IN PROPELLER

(1) DRILLED BOLT WITH CASTELATED NUT OR UNDRILLED BOLT WITH SELF-LOCKING NUT.

(2) BOLT WITH HEAD DRILLED FOR SAFETY WIRING.

NOTE: THESE REPAIRS ARE PERMITTED ONLY ON THE DRIVING FLANGE OF THE PROPELLER HUB AND THE ADJACENT FACE OF THE PROPELLER.

FIGURE 14-8. Repair of Hub and Propeller With Elongated or Damaged Bolt Holes. Ref. CAM 18.20-14(f)(5), 18.20-14(f)(6), 18.20-14(f)(6)(i), 18.20-14(f)(6)(ii)

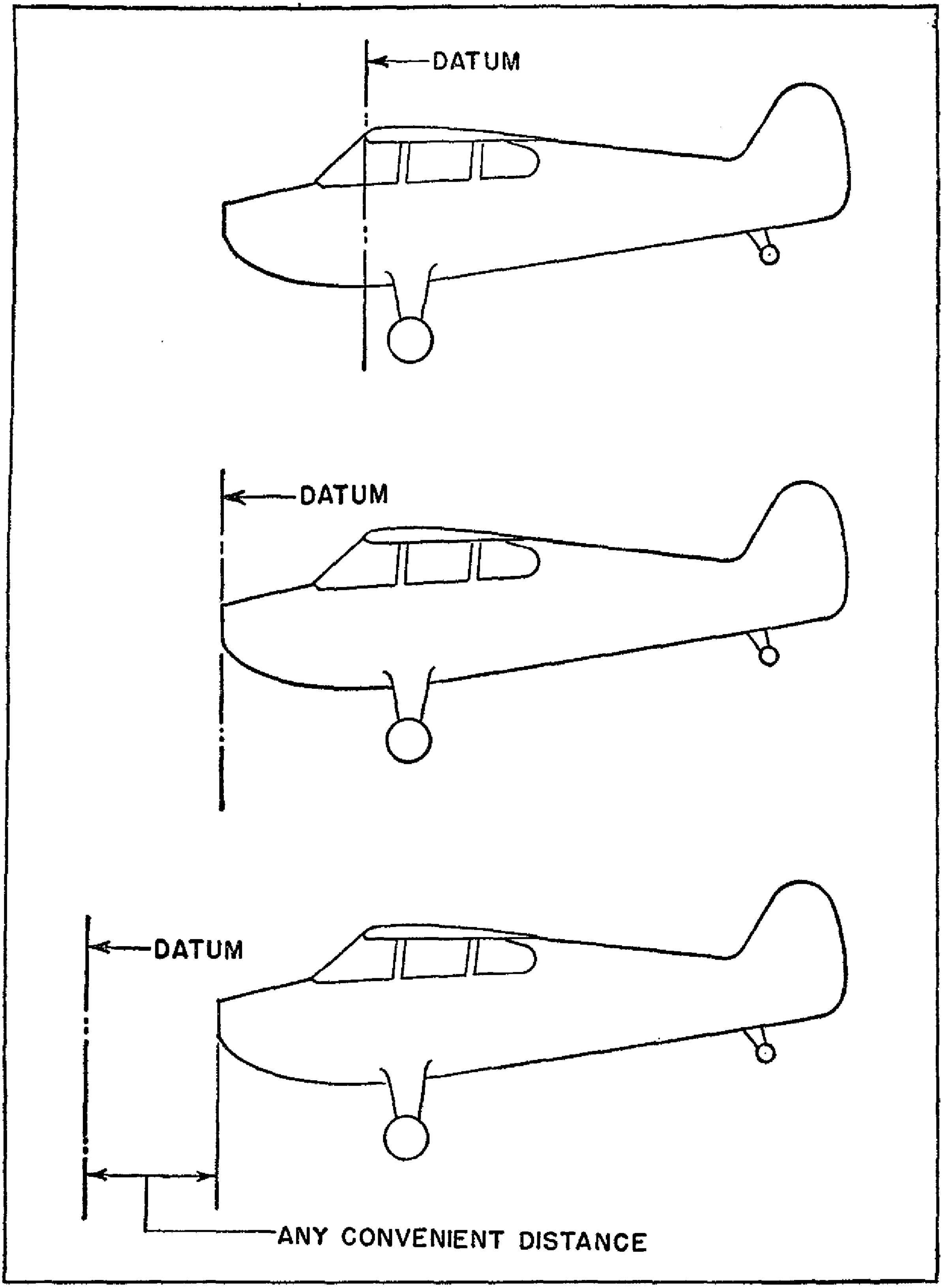


FIGURE 15–1. Typical Datum Locations. Ref. CAM 18.20–15(a)(1)(v)

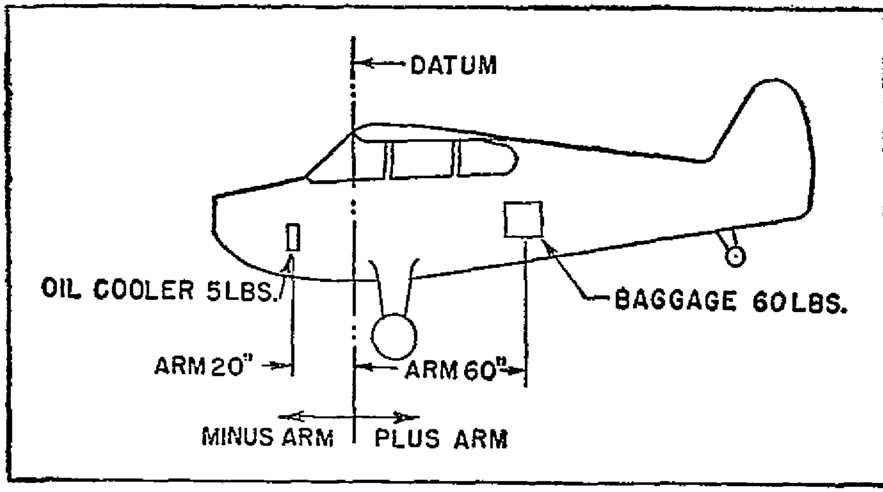
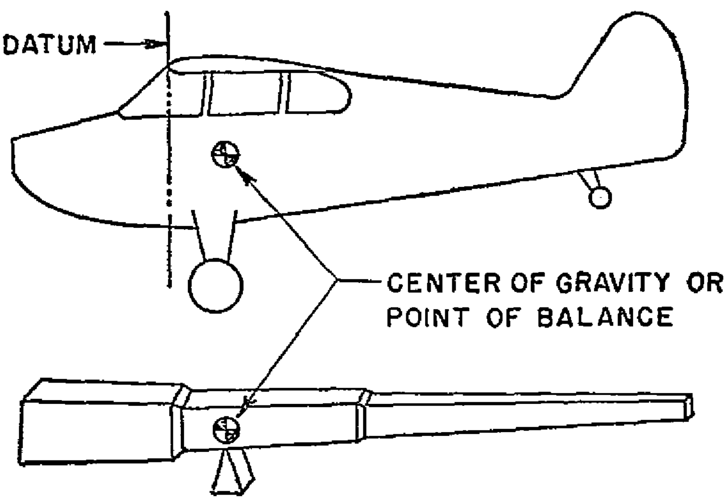


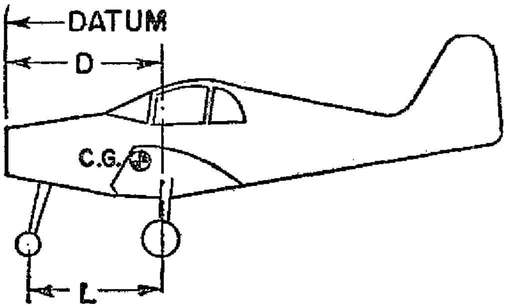
FIGURE 15-2. Illustration of Arm (or Moment Arm). Ref. CAM 18.20-15(a)(1)(vi)



The entire aircraft weight may be considered to be concentrated at the center of gravity. Therefore, the moment of the aircraft about the datum is the weight of the aircraft times the horizontal distance between the C.G. and the datum.

Example: If the weight of this airplane is 2000 lbs. and the arm from the datum to the center of gravity is 16 inches, the moment of the aircraft about the datum is 2000 x 16 or 32,000 in. lbs.

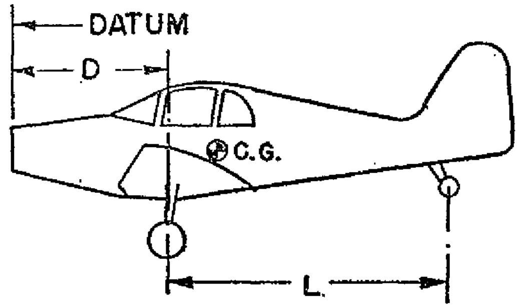
FIGURE 15-3. Example of Moment Computation. Ref. CAM 18.20-15(a)(1)(vii)



NOSE WHEEL TYPE AIRCRAFT

DATUM LOCATED FORWARD OF THE MAIN WHEELS

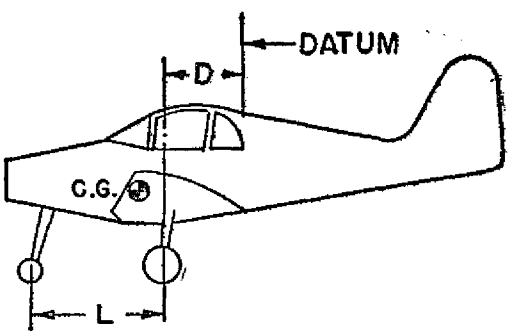
$$C.G. = D - \left(\frac{F \times L}{W}\right)$$



TAIL WHEEL TYPE AIRCRAFT

DATUM LOCATED FORWARD OF THE MAIN WHEELS

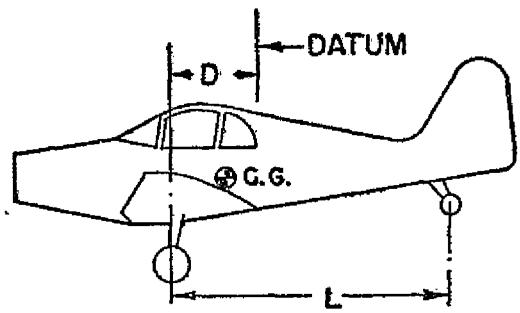
$$C.G. = D + \left(\frac{R \times L}{W}\right)$$



NOSE WHEEL TYPE AIRCRAFT

DATUM LOCATED AFT OF THE MAIN WHEELS

$$C.G. = -\left(D + \frac{F \times L}{W}\right)$$



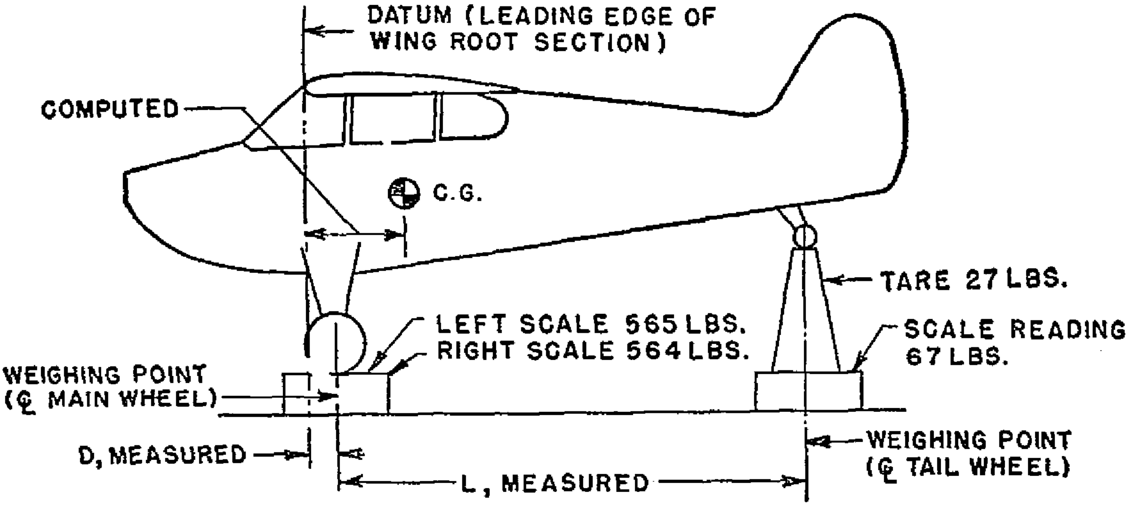
TAIL WHEEL TYPE AIRCRAFT

DATUM LOCATED AFT OF THE MAIN WHEELS

$$C.G. = -D + \left(\frac{R \times L}{W}\right)$$

CG = Distance from datum to center of gravity of the aircraft.
W = The weight of the aircraft at the time of weighing.
D = The horizontal distance measured from the datum to the main wheel weighing point.
L = The horizontal distance measured from the main wheel weighing point to the nose or tail weighing point.
F = The weight at the nose weighing point.
R = The weight at the tail weighing point.

FIGURE 15–4. Empty Weight Center-of-Gravity Formulas. Ref. CAM 18.20–15(a)(1)(ix)



TO FIND: EMPTY WEIGHT AND EMPTY WEIGHT CENTER OF GRAVITY

Datum is the leading edge of the wing (from aircraft specification)
(D) Actual measured horizontal distance from the main wheel weighing point ℄ main wheel) to the Datum---3"
(L) Actual measured horizontal distance from the rear wheel weighing point (℄ rear wheel) to the main wheel weighing point--------------------------------------222"

SOLVING: EMPTY WEIGHT

Weighing Point	Scale Reading #	Tare #	Net Weight #
Right	564	0	564
Left	565	0	565
Rear	67	27	40
Empty Weight (W)			1169

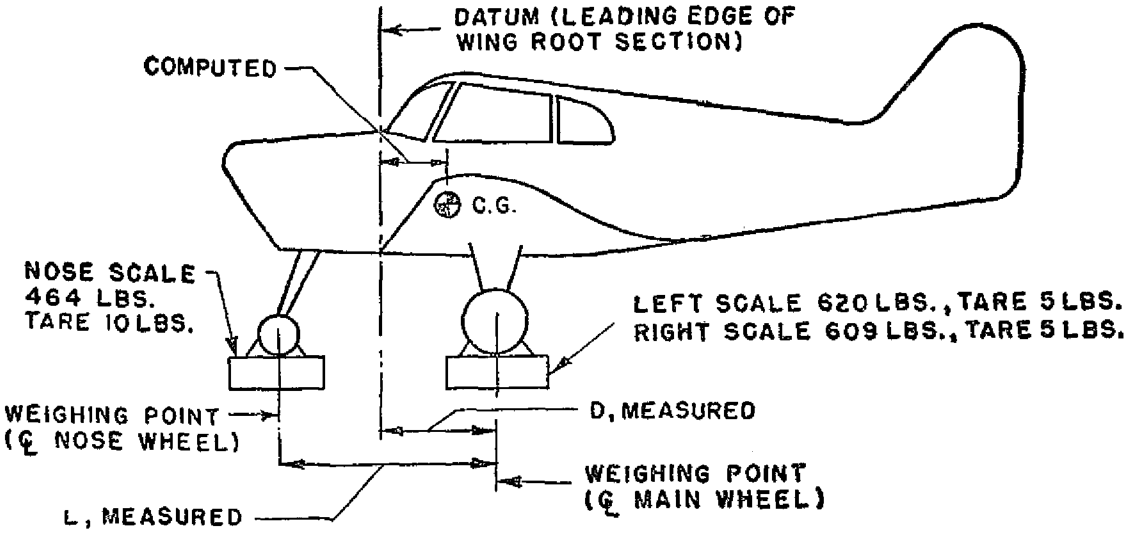
SOLVING: EMPTY WEIGHT CENTER OF GRAVITY

Formula: $\text{C.G.} = D + \frac{R \times L}{W} = 3'' + \frac{40 \times 222}{1169} = 3'' + 7.6'' = 10.6''$

Reference for formula Fig. 15-4

This case is shown properly entered on a sample weight and balance report form, Figure 15-17.

FIGURE 15-5. Empty Weight and Empty Weight Center-of-Gravity—Tail-Wheel Type Aircraft. Ref. CAM 18.20-15(a)(1)(ix) and CAM 18.20-15(c)(6)



TO FIND: EMPTY WEIGHT AND EMPTY WEIGHT CENTER OF GRAVITY

Datum is the leading edge of the wing (from aircraft specification)

(D) Actual measured horizontal distance from the main wheel weight point (℄ main wheel) to the Datum-- 34.0''

(L) Actual measured horizontal distance from the front wheel weighing point (℄ front wheel) to the main wheel weighing point---------------------------------- 67.8''

SOLVING: EMPTY WEIGHT

Weighing Point	Scale Reading #	Tare #	Net Weight
Right	609	5	604
Left	620	5	615
Front	464	10	454
Empty Weight (W)			1673

SOLVING: EMPTY WEIGHT CENTER OF GRAVITY

Formula: $C.G. = D - \frac{F \times L}{W} = 34'' - \frac{454 \times 67.8}{1673} = 34'' - 18.3'' = 15.7''$

Reference for formula Fig. 15-4.

FIGURE 15-6. Empty Weight and Empty Weight Center-of-Gravity—Nose-Wheel Type Aircraft. Ref. CAM 18.20-15(a)(1)(ix)

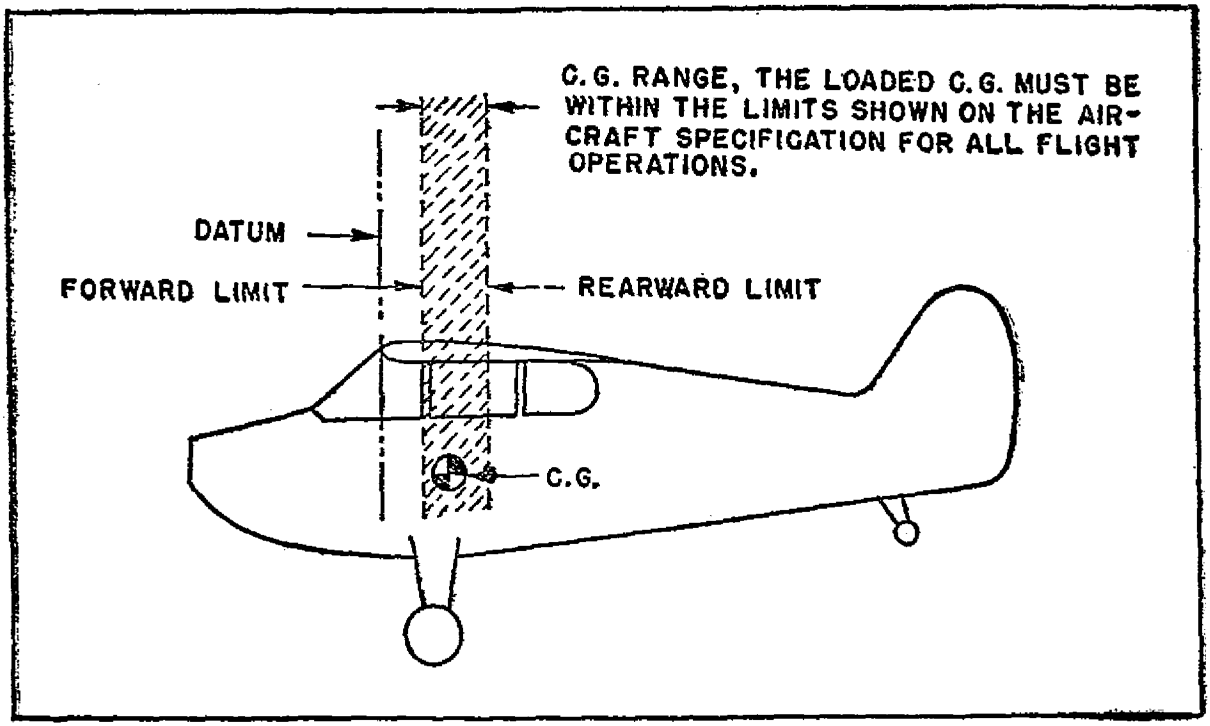


FIGURE 15–7. Operating Center-of-Gravity Range. Ref. CAM 18.20–15(a)(1)(xi)

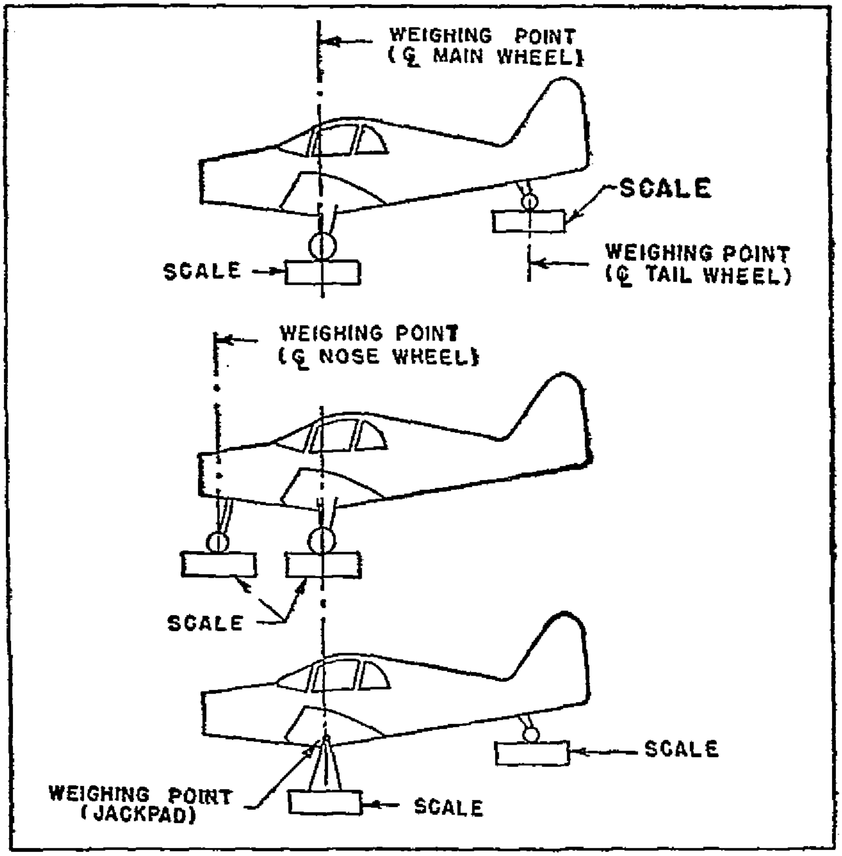
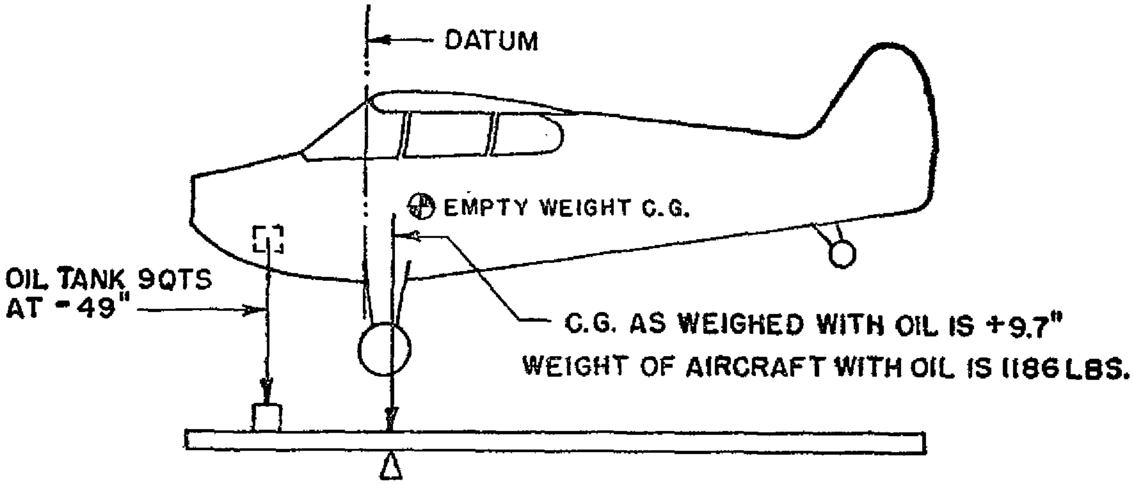


FIGURE 15–8. Weighing Point Centerline. Ref. CAM 18.20–15(a)(1)(xiii)



EMPTY WEIGHT AND EMPTY WEIGHT CENTER OF GRAVITY
(when aircraft is weighed with oil)

GIVEN:

Aircraft as weighed with full oil------ 1186 lbs.
Center of gravity---------------------- 9.7"
Full oil capacity 9 qts.----------------- 17 lbs.

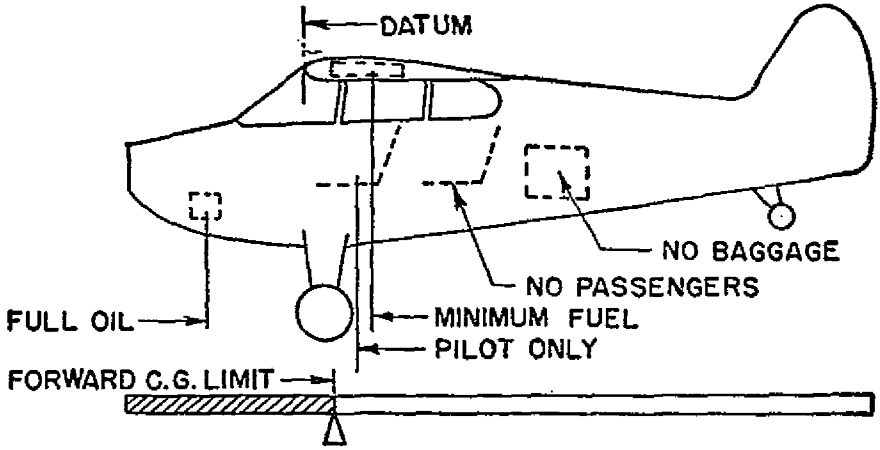
SOLVING:

	Weight #	x Arm"	Moment "#
Aircraft as weighed	+ 1186	+ 9.7	+ 11504
Less oil	- 17	- 49.0	+ 833
Total	+ 1169(A)		+ 12337(B)

Empty Weight (A) = 1169 pounds.

Empty Weight Center of Gravity $\frac{B}{A} = \frac{12337}{1169} = +10.6''$

FIGURE 15-9. Empty Weight and Empty Weight Center-of-Gravity When Aircraft Is Weighed With Oil. Ref. CAM 18.20-15(b)(7)



TO CHECK: MOST FORWARD WEIGHT AND BALANCE EXTREME.

GIVEN: Actual empty weight of the airplane -------------- 1169#
Empty weight center of gravity ----------------- +10.6"
*Maximum weight ---------------------------- 2100#
*Forward C.G. limit ------------------------- + 8.5"
*Oil, capacity 9 qts. ------------------------- 17# at - 49
*Pilot in farthest forward seat equipped with controls (unless otherwise placarded) ----------- 170# at + 16"
*Since the fuel tank is located to the rear of the forward C.G. limit, minimum fuel should be included. $\frac{\text{METO HP}}{12} = \frac{165}{12} =$ 13.75 gal. x 6# ------- 83# at + 22"

*Information should be obtained from the aircraft specification.

Note: Any items or passengers must be used if they are located ahead of the forward C.G. limit.
Full fuel must be used if the tank is located ahead of the forward C.G. limit.

CHECK OF FORWARD WEIGHT AND BALANCE EXTREME

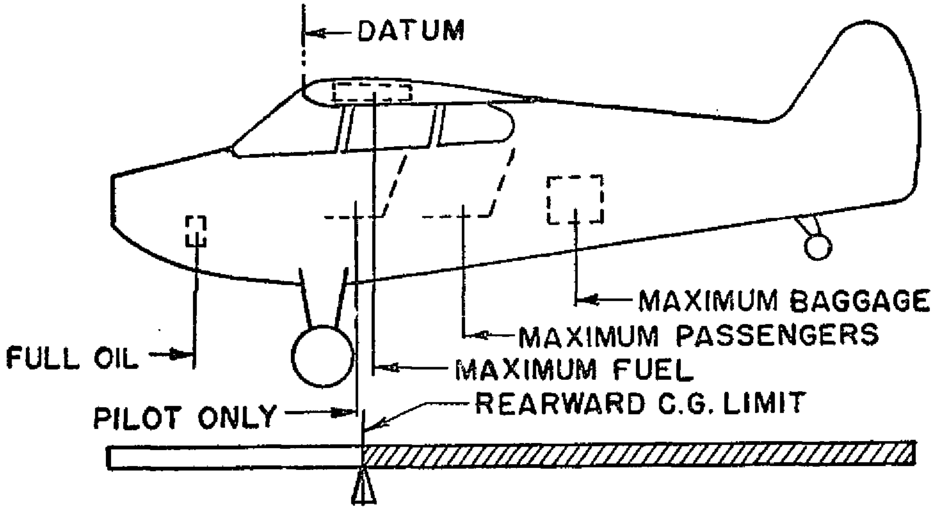
	Weight (#)	x Arm (")	Moment ("#)
Aircraft Empty	+ 1169	+ 10.6	+ 12391
Oil	+ 17	- 49	- 833
Pilot	+ 170	+ 16	+ 2720
Fuel	+ 83	+ 22	+ 1826
Total	+ 1439 (TW)		+ 16104 (TM)

Divide the TM (Total Moment) by the TW (Total Weight) to obtain the forward weight and balance extreme.

$$\frac{TM}{TW} = \frac{16104}{1439} = +11.2''$$

Since the forward C.G. limit and the maximum weight are not exceeded, the forward weight and balance extreme condition is satisfactory.

FIGURE 15-10. Example of Check of Most Forward Weight and Balance Extreme. Ref. CAM 18.20-15(c)(3)(i)(*c*) and CAM 18.20-15(c)(6)



TO CHECK: MOST REARWARD WEIGHT AND BALANCE EXTREME.

GIVEN: Actual empty weight of the airplane --------- 1169#
Empty weight center of gravity ------------- 10.6"
*Maximum weight ------------------------ 2100#
*Rearward C.G. limit --------------------- 21.9"
*Oil capacity 9 qts. ---------------------- 17# at - 49"
*Baggage, placarded do not exceed 100 lbs. --- 100# at + 75.5"
*Two passengers in rear seats, 170 x 2 ------ 340# at + 48"
*Pilot in most rearward seat equipped with controls (unless otherwise placarded) ------- 170# at + 16"
*Since the fuel tank is located aft of the rearward C.G. limit full fuel must be used --- 240# at + 22"

* Information should be obtained from the aircraft specification.

Note: If fuel tanks are located ahead of the rearward C.G. limit minimum fuel should be used.

CHECK OF REARWARD WEIGHT AND BALANCE EXTREME

	Weight (#)	x Arm (")	= Moment ("#)
Aircraft empty	+ 1169	+ 10.6	+ 12391
Oil	+ 17	- 49	- 833
Pilot (1)	+ 170	+ 16	+ 2720
Passengers (2)	+ 340	+ 48	+ 16320
Fuel (40 gals.)	+ 240	+ 22	+ 5280
Baggage	+ 100	+ 75.5	+ 7550
Total	+2036 (TW)		+ 43428 (TM)

Divide the TM (Total Moment) by the TW (Total Weight) to obtain the rearward weight and balance extreme.

$$\frac{TM}{TW} = \frac{43428}{2036} = + 21.3''$$

Since the rearward C.G. limit and the maximum weight are not exceeded, the rearward weight and balance extreme condition is satisfactory.

FIGURE 15–11. Example of Check of Most Rearward Weight and Balance Extreme. Ref. CAM 18.20-15(c)(3)(ii)(c) and CAM 18.20-15(c)(6)

EXAMPLE OF THE DETERMINATION OF THE NUMBER OF PASSENGERS AND BAGGAGE PERMISSIBLE WITH FULL FUEL

GIVEN:

Actual empty weight of the aircraft ------------------ 1169#
Empty weight center of gravity ---------------------- 10.6"
Maximum weight ------------------------------------- 2100#
Datum is leading edge of the wing
Forward center of gravity limit --------------------- 8.5"
Rearward center of gravity limit -------------------- 21.9"
Oil capacity, 9 qts.; show full capacity ---------------- 17# at -49"
Baggage, maximum ----------------------------------- 100# at +75.5"
Two passengers in rear seat, 170# x 2 ---------------- 340# at +48"
Pilot in most rearward seat equipped with
controls (unless otherwise placarded) ---------------- 170# at +16"
Full fuel, 40 gals. x 6# ----------------------------- 240# at +22"

	Weight(#)	x Arm(")	= Moment("#)
Aircraft empty	+ 1169	+ 10.6	+ 12391
Oil	+ 17	- 49	- 833
Full fuel	+ 240	+ 22	+ 5280
Passengers, 2 rear	+ 340 *	+ 48	+ 16320
Pilot	+ 170	+ 16	+ 2720
Baggage	+ 100	+ 75.5	+ 7550
Total	+ 2036 (TW)		+ 43428 (TM)

Divide the TM. (total moment) by the TW (total weight) to obtain the loaded center of gravity.

$$\frac{TM}{TW} = \frac{43428}{2036} = +21.3''$$

The above computations show that with full fuel, 100 pounds of baggage and two passengers in the rear seat may be carried in this aircraft without exceeding either the maximum weight or the approved C.G. range.

This condition may be entered in the loading schedule as follows:

GALLONS OF FUEL	NUMBER OF PASSENGERS	POUNDS OF BAGGAGE
Full	2 Rear	100

* Only two passengers are listed to prevent the maximum weight of 2100 lbs. from being exceeded.

FIGURE 15-12. Loading Conditions: Determination of the Number of Passengers and Baggage Permissible With Full Fuel. Ref. CAM 18.20-15(c)(4)

EXAMPLE OF THE DETERMINATION OF THE POUNDS OF FUEL AND BAGGAGE PERMISSIBLE WITH MAXIMUM PASSENGERS

	Weight (#)	x Arm (")	= Moment ("#)
Aircraft empty	+ 1169	+ 10.6	+ 12391
Oil	+ 17	- 49	- 833
Pilot	+ 170	+ 16	+ 2720
Passenger, 1 front	+ 170	+ 16	+ 2720
Passengers, 2 rear	+ 340	+ 48	+ 16320
Fuel (39 gals.)	+ 234	+ 22	+ 5148
Baggage	---	---	---
Total	+ 2100		+ 38466

Divide the TM (total moment) by the TW (total weight) to obtain the loaded center of gravity.

$$\frac{TM}{TW} = \frac{38466}{2100} = +\ 18.3''$$

The above computations show that with the maximum number of passengers, 39 gallons of fuel and zero pounds of baggage may be carried in this aircraft without exceeding either the maximum weight or the approved C.G. range.

This condition may be entered in the loading schedule as follows:

GALLONS OF FUEL	NUMBER OF PASSENGERS	POUNDS OF BAGGAGE
* FULL	*2 rear	* 100
39	1(F) 2(R)	None

* Conditions as entered from Figure 15-12.
(F) Front seat
(R) Rear seat

FIGURE 15-13. Loading Conditions: Determination of the Fuel and Baggage Permissible With Maximum Passengers. Ref. CAM 18.20-15(c) (4)

EXAMPLE OF THE DETERMINATION OF THE FUEL AND THE NUMBER AND LOCATION OF PASSENGERS PERMISSIBLE WITH MAXIMUM BAGGAGE

	Weight (#)	x Arm ('')	= Moment (''#)
Aircraft empty	+ 1169	+ 10.6	+ 12391
Oil	+ 17	- 49	- 833
Pilot	+ 170	+ 16	+ 2720
Passenger (1) rear	+ 170	+ 48	+ 8160
Passenger (1) front	+ 170	+ 16	+ 2720
Fuel (40 gals.)	+ 240	+ 22	+ 5280
Baggage	+ 100	+ 75.5	+ 7550
Total	+ 2036		+ 37988

Divide the TM (total moment) by the TW (total weight) to obtain the loaded center of gravity.

$$\frac{TM}{TW} = \frac{37988}{2036} = +18.7$$

The above computations show that with maximum baggage, full fuel and 2 passengers (1 in the front seat and 1 in the rear seat) may be carried in this aircraft without exceeding either the maximum weight or the approved C.G. range.

This condition may be entered in the loading schedule as follows:

GALLONS OF FUEL	NUMBER OF PASSENGERS	POUNDS OF BAGGAGE
* Full	* 2 Rear	*100
** 39	*1(F) 2(R)	**None
Full	1(F) 1(R)	Full

*Conditions as entered from Figure 15-12
**Conditions as entered from Figure 15-13
(F) Front seat
(R) Rear seat

FIGURE 15-14. Loading Conditions: Determination of the Fuel and the Number and Location of Passengers Permissible With Maximum Baggage. Ref. CAM 18.20-15(c)(4)

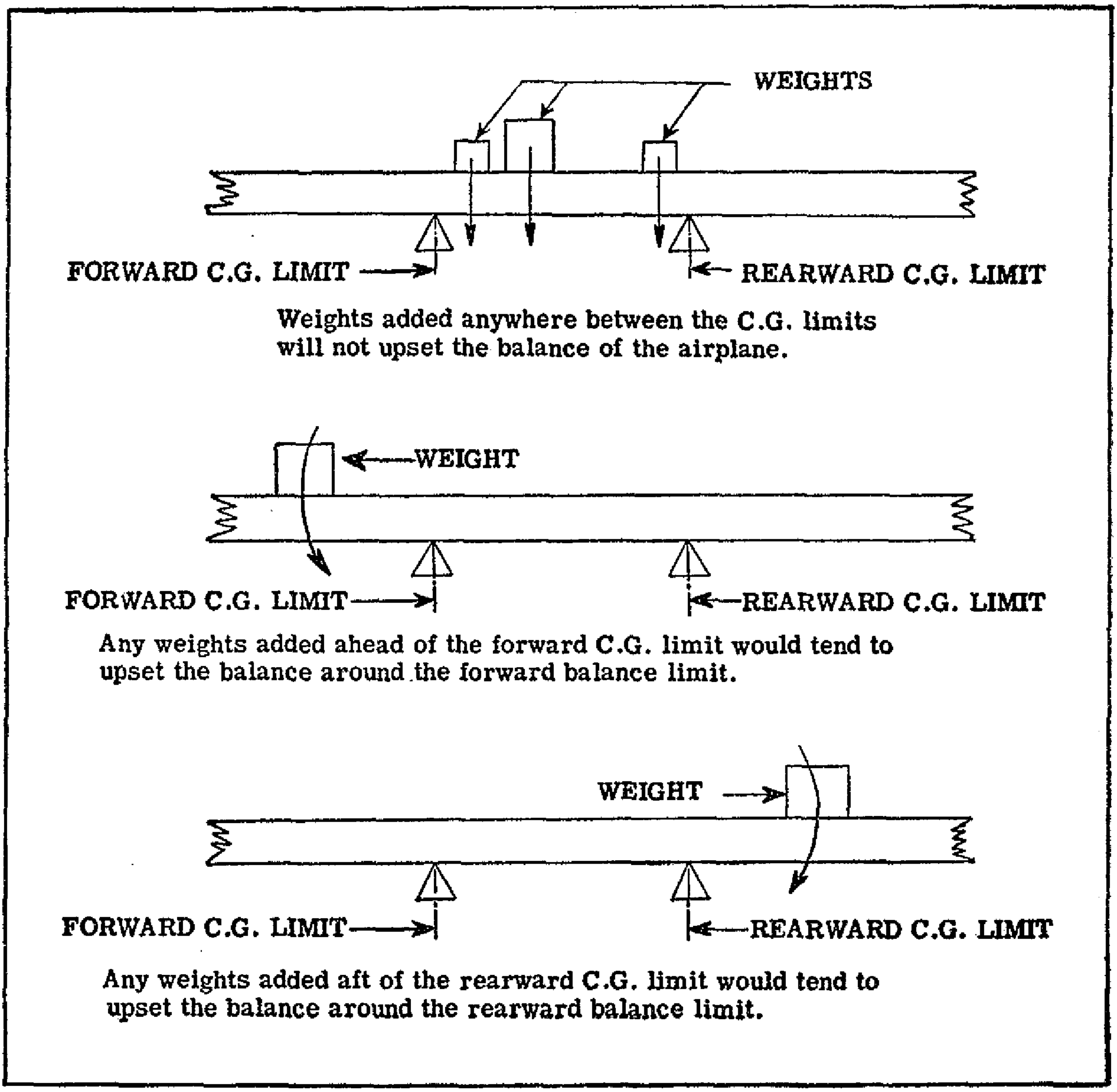
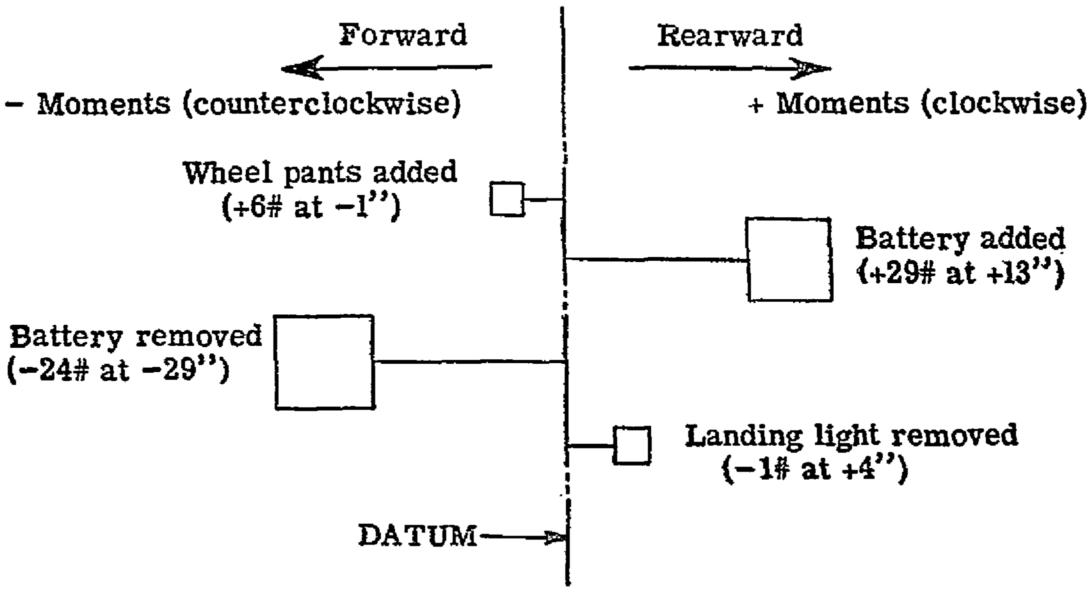


FIGURE 15-15. Effects of the Addition of Equipment Items on Balance. Ref. CAM 18.20-15(c) (5)(i)



	Weight (#) x	Arm (") =	Moment ("#)
ADDED			
Item 204 wheel pants	+ 6	– 1	– 6
Item 302b Battery	+ 29	+13	+377
REMOVED			
Item 302a Battery	– 24	–29	+696
Item 303 Landing light	– 1	+ 4	– 4

This condition is shown properly entered on a sample weight and balance report on Figure 15-18 under Equipment Change.

FIGURE 15–16. Example of Moment and Weight Changes Resulting from Equipment Changes. Ref. CAM 18.20–15(c)(5)(i)

MAKE MA-700 MODEL A' SERIAL # 0000 REGISTRATION # N1234

DATUM IS leading edge of wing.

COMPUTE AS FOLLOWS IF AIRCRAFT WEIGHED

1. Leveling means: level top longeron between front and rear seats.
2. Main wheel weighing point is located (______"FORWARD)(+ 3 " AFT) of datum.
3. Actual measured distance from the main weight point centerline to the tail (or nose) point centerline 222 ".
4. Oil over and above "ZERO" tank reading = (a. — Gals.)(b. — Lbs.)(c. — In.)

ACTUAL EMPTY WEIGHT

	Weight Point	Scale Reading -	Tare =	Net Weight
5.	Right	564	0	564
6.	Left	565	0	565
7.	Tail	67	27	40
8.	Nose	—	—	—
9.	Total Net Weight			1169

CENTER OF GRAVITY AS WEIGHED

10. C.G. relative to main wheel weighing point:

(a) Tail wheel airc. $\frac{\text{(item 3, 222)} \quad \text{X} \quad \text{(Item 7, 40)}}{\text{(Item 9, 1169)}}$ = + 7.6 = C.G.

(b) Nose wheel airc. $\frac{\text{(Item 3 —)} \quad \text{X} \quad \text{(Item 8 —)}}{\text{(Item 9 —)}}$ = ______ = C.G.

11. C.G. relative to datum:

(a) Tail wheel airc. (Item 10a, + 7.6) added to (Item 2, + 3) = + 10.6" = C.G.

(b) Nose wheel airc. (Item 10b, ______) added to (Item 2, ______) = ______ = C.G.

COMPUTE IF AIRCRAFT WEIGHED WITH OIL (Item 4)

	Weight X	Arm =	Moment
Aircraft	(9)	(11)	
Less Oil	(4b)	(4c)	
Empty Totals	(a)		(b)

12. $\frac{\text{(b)}\text{———————}}{\text{(a)}}$ = (c)——————" = Empty weight C.G.

REPAIR AGENCY______________________ ______________ DATE______________

Name Number

FIGURE 15-17. Sample Weight and Balance Report to Determine Empty Weight Center-of-Gravity, Ref. CAM 18.20-15(c)(6)

EQUIPMENT LIST

*Required or Optional Item Numbers as Shown in Aircraft Specification						
1	2	101	102	103	104	105
106	201	202	203	301	302(a)	303
401(a)	402	—	—	—	—	—

Special Equipment				
Item	Make	Model	Weight	Arm
3 Flares 1½ Min.	XYZ	03	25#	150"
Enter above those items included in the empty weight.				

WEIGHT AND BALANCE EXTREME CONDITIONS

Approved fwd limit 8.5" Approved max. weight 2100# Approved aft limit 21.9"

Item	FORWARD CHECK			REARWARD CHECK		
	Weight X	Arm =	Moment	Weight X	Arm =	Moment
Airc. Empty	+ 1169 (9 or 12a)	+ 10.6 (11 or 12c)	+ 12391	+ 1169 (9 or 12a)	+ 10.6 (11 or 12c)	+ 12391
Oil	+ 17	- 49	- 833	+ 17	- 49	- 833
Pilot	+ 170	+ 16	+ 2720	+ 170	+ 16	+ 2720
Fuel	+ 83	+ 22	+ 1826	+ 240	+ 22	+ 5280
Passenger(s)				+ 340	+ 48	+ 16320
Baggage				+ 100	+ 75.5	+ 7550
TOTAL	+ 1439= TW		+ 16104= TM	+ 2036=TW		+ 43428=TM

$\frac{TM}{TW} = \frac{16104}{1439} = +11.2'' =$ Most Forward C.G. location

$\frac{TM}{TW} = \frac{43428}{2036} = +21.3'' =$ Most rearward C.G. location

LOADING SCHEDULE

Gallons of Fuel	Number of Passengers	Pounds of Baggage
40	2(R)	100
The above includes pilot and capacity oil.		

EQUIPMENT CHANGE

Computing New C.G.			
Item, Make, and Model*	Weight X	Arm =	Moment
Airc. Empty	+ 1169 (9 or 12a)	+ 10.6 (11 or 12c)	+ 12391
204 added	+ 6	- 1	- 6
302(b) added	+ 29	+ 13	+ 377
302(a) removed	- 24	- 29	+ 696
303 removed	- 1	+ 4	- 4
NET TOTALS	- 1179 = NW		+ 13454 = NM

$\frac{NM}{NW} = \frac{13454}{1179} = +11.4'' =$ New C.G.

*ITEM NUMBERS WHEN LISTED IN PERTINENT AIRCRAFT SPECIFICATION MAY BE USED IN LIEU OF "ITEM, MAKE, AND MODEL".

APPROVED BY ______________ DATE ______________

FIGURE 15-18. Sample Weight and Balance Report Including an Equipment Change for Aircraft Fully Loaded. Ref. CAM 18.20-15(c)(5)(i) and CAM 18.20-15(c)(6)

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