Advisory Circular

Subject: Acceptable Methods, Techniques, and Practices—Aircraft Alterations

Date: [Type the date here.]

Initiated by: AFS-306

AC No: 43.13-2B

1. PURPOSE. This advisory circular (AC) contains methods, techniques, and practices acceptable to the Administrator for the inspection and alteration on non-pressurized areas of civil aircraft of 12,500 lbs gross weight or less. This data generally pertains to minor alterations; however, the alteration data herein may be used as approved data for major alterations when the AC chapter, page, and paragraph are listed in block 8 of FAA form 337 when the user has determined that it is:

a. Appropriate to the product being altered,

b. Directly applicable to the alteration being made, and

c. Not contrary to manufacturer's data.

2. CANCELLATION. AC 43.13-2A, Acceptable Methods, Techniques, and Practices—Aircraft Alterations, dated 1977, is canceled.

3. REFERENCE. Title 14 of the Code of Federal regulations part 43, § 43.13(a) states that each person performing maintenance, alteration, or preventive maintenance on an aircraft, engine, propeller, or appliance shall use the methods, techniques, and practices prescribed in the current manufacturer's maintenance manual or Instructions for Continued Airworthiness prepared by its manufacturer, or other methods, techniques, or practices acceptable to the Administrator, except as noted in § 43.16. FAA Inspectors are prepared to answer questions that may arise in this regard. Persons engaged in the inspection and alteration of civil aircraft should be familiar with 14 CFR part 43, Maintenance, Preventive Maintenance, Rebuilding, and Alterations, and part 65, subparts A, D, and E of Certification: Airmen Other than Flight Crewmembers, and applicable airworthiness requirements under which the aircraft was type certificated.

4. COMMENTS INVITED. Comments regarding this AC should be directed to DOT/FAA: ATTN: Bill O'Brien, Independence Av, Washington, DC 20591, FAX 202 267-5115 or william.o'brien@faa.gov.

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CHAPTER 1. STRUCTURAL DATA

100. GENERAL. Structural integrity is a major factor in aircraft design and construction. Addition or removal of equipment involving changes in weight could affect the structural integrity, weight, balance, flight characteristics, or performance of an aircraft.

101. STRUCTURAL DESIGN PROCESS. Structural design processes follows these steps:

a. Determine the overall load factors.

b. Estimate the resulting loads.

c. Distribute these loads over the aircraft.

d. Determine the material, size, and shape of the part.

e. Calculate the resulting stresses in the part.

f. Compare these stresses with the maximum allowable for the material used.

g. Resize the part as necessary.

102. TYPES OF LOADS.

a. Aircraft parts have to be formed out of different types of material and joined together. Each of those parts carries a load and the fastener that brings these parts together has to carry the load from one part to the other.

b. Every aircraft is subject to structural stress. Stress acts on an aircraft whether it is on the ground or in flight. Stress is defined as a load applied to a unit area of material.

c. Tension is the stress acting against another force that is trying to pull something apart.

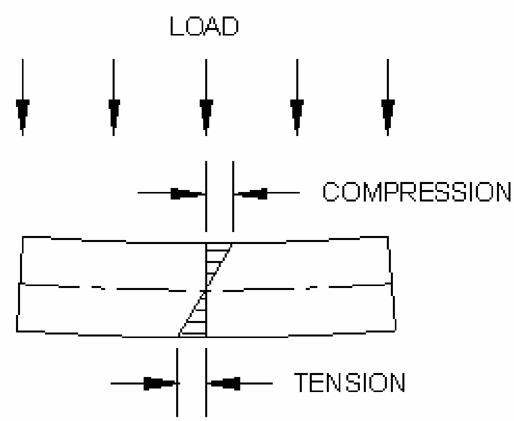
d. Compression is a squeezing or crushing force that tries to make parts smaller.

e. Torsion is a twisting force.

f. Shear stress tends to slide one piece of material over another.

g. Bending is a combination of two forces, compression, and tension. During bending stress, the material on the inside of the bend is compressed and the outside material is stretched in tension.

FIGURE 1-1. BENDING OF A BEAM



h. In flight, an aircraft structure in flight is subjected to variable stresses due to the varying loads that may be imposed. The designer's problem involves anticipating the possible stresses that the structure will have to endure and build the structure strong enough to withstand these stresses.

103. STATIC LOADS. Static loads are loads which do not undergo change in magnitude or direction during a measurement procedure. Load factors are defined as follows:

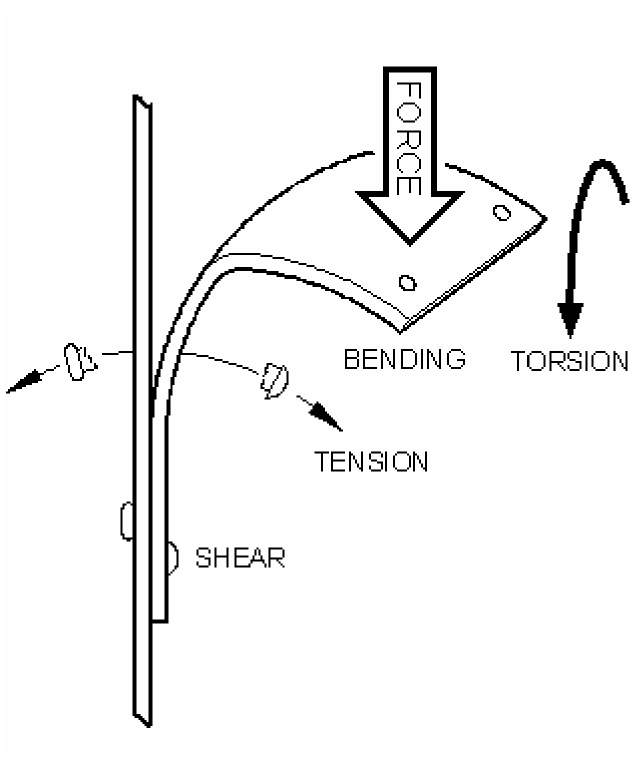
a. Limit load factors are the maximum load factors which may be expected during service (the maneuvering, gust, or ground load factors established by the manufacturer for type certification).

b. Ultimate load factors are the limit load factors multiplied by a prescribed factor of safety. Certain loads, such as the minimum ultimate inertia forces prescribed for emergency landing conditions, are given directly in terms of ultimate loads.

c. Static test load factors are the ultimate load factors multiplied by prescribed casting, fitting, bearing, and/or other special factors. Where no special factors apply, the static test load factors are equal to the ultimate load factors.

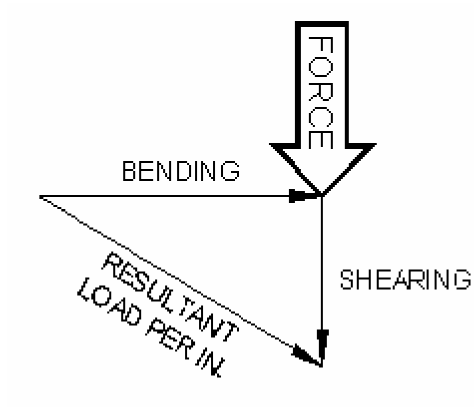
d. Critical Static Test Load Factors are the greater of the maneuvering, gust, ground, and inertia load static test load factors for each direction (up, down, starboard, port, fore, and aft).

FIGURE 1-2. TYPICAL LOAD FACTORS



104. LOAD ANALYSIS. Calculate load factors for your aircraft per Title 14 of the Code of Federal Regulations (14 CFR) part 23: construct Vn diagram, calculate air load distributions, and landing loads.

FIGURE 1-3. RESULTANT LOAD



105. STRUCTURAL SIZING AND ANALYSIS. Design and size your load structures, including wing spars, wing attach fittings, stabilizers, landing gear struts, etc.

a. Static tests using the following load factors are acceptable for equipment installations:

TABLE 1-1. LIMIT LOAD FACTORS

Direction of Force Applied	Normal-Utility 14 CFR part 23 (CAR 3)	Acrobatic 14 CFR part 23 (CAR 3)	Transport 14 CFR part 25 (CAR 4b)	Rotorcraft 14 CFR part 27, 29 (CAR 6, 7)
Sideward	1.5g	1.5g	1.5g	2.0g
Upward	3.0g	4.5g	* *	1.5g
Forward	9.0g	9.0g	9.0g	4.0g
Downward	6.6g	9.0g	* *	4.0g

*When equipment mounting is located externally to one side, or forward of occupants, a forward load factor of 2.0g is sufficient.
**Due to differences among various aircraft designs in flight and ground load factors, contact the aircraft manufacturer for the load factors required for a given model and location. In lieu of specific information, the factors used for part 23 utility category are acceptable for aircraft that never exceed speeds of 250 knots or less and the factors used for part 23 acrobatic category for all other transport aircraft.

b. The following is an example of determining the static test loads for a 7-pound piece of equipment to be installed in a utility category aircraft (part 23).

TABLE 1-2. SAMPLE LOAD FACTORS

Load Factors (From the above table)	Static Test Loads (Load factor X 7 pounds)
Sideward 1.5g	10.5 pounds
Upward 3.0g	21.0 pounds
Downward 6.6g	46.2 pounds
Forward 9.0g	63.0 pounds

c. When an additional load is to be added to structure already supporting previously installed equipment, determine the capability of the structure to support the total load (previous load plus added load).

106. STATIC TESTS.

CAUTION: The aircraft and/or equipment can be damaged in applying static loads, particularly if a careless or improper procedure is used. It is recommended, whenever practicable, that static testing be conducted on a duplicate installation in a jig or mockup which simulates the related aircraft structure. Static test loads may exceed the yield limits of the assemblies being substantiated and can result in partially sheared fasteners, elongated holes, or other damage which may not be visible unless the structure is disassembled. If the structure is materially weakened during testing, it may fail at a later date. Riveted sheet metal and composite laminate construction methods especially do not lend themselves to easy detection of such damage. To conduct static tests:

a. Determine the weight and center of gravity position of the equipment item.

b. Make actual or simulated installation of attachment in the aircraft or preferably on a jig using the applicable static test load factors.

c. Determine the critical ultimate load factors for the up, down, starboard, port, fore, and aft directions. A hypothetical which follows steps (1) through (4) below pertains to the example in Figure 1-4, Hypothetical of Determining Static Test Loads.

(1) Convert the gust, maneuvering, and ground load factors obtained from the manufacturer or Federal Aviation Administration (FAA) engineer to determine the ultimate load factors. Unless otherwise specified in the airworthiness standards applicable to the aircraft, ultimate load factors are limit load factors multiplied by a 1.5 safety factor. (See columns 1, 2, and 3 for items A, B, and C.)

(2) Determine the ultimate inertia load forces for the emergency landing conditions as prescribed in the applicable airworthiness standards. (See items D and E, column 3.)

(3) Determine what additional load factors are applicable to the specific seat, litter, berth, or cargo tiedown device installation. The ultimate load factors are then multiplied by these factors to obtain the static test factors. (To simplify this example, only the seat, litter, berth, and safety belt attachment factor of 1.33 was assumed to be applicable. See item E, column 4.)

(4) Select the highest static test load factors obtained in steps 1, 2, and/or 3 for each direction (up, down, starboard, port, fore, and aft). These factors are the critical static test load factors used to compute the static test load. (See Column 6.)

d. Apply a load at center of gravity position (of equipment item or dummy) by any suitable means to demonstrate that the attachment and structure are capable of supporting the required loads. When no damage or permanent deformation occurs after 3 seconds of applied static load, the structure and attachments are acceptable. Should permanent deformation occur after 3 seconds, repair or replace the deformed structure to return it to its normal configuration and strength. Additional load testing is not necessary.

107. MATERIALS AND WORKMANSHIP. Use materials conforming to an accepted standard such as Army/Navy and Airforce/Navy (AN) National Aerospace Standards (NAS), Technical Standard Order (TSO), or Military Specifications (MIL-SPEC).

a. Suitability and durability of materials used for parts, the failure of which could adversely affect safety, must:

(1) Be established by experience or tests;

(2) Meet approved specifications that ensure the strength and other properties, assumed in the design data; and

(3) Take into account the effects of environmental conditions, such as temperature and humidity, expected in service.

b. Workmanship must be of a high standard.

108. MATERIAL STRENGTH PROPERTIES AND DESIGN VALUES. Material strength properties must be based on enough tests of material meeting specifications to establish design values on a statistical basis. Design values must be chosen to minimize the probability of structural failure due to material variability. Except as provided in subparagraph e below, compliance with this paragraph must be shown by selecting design values that ensure material strength with the following probabilities:

a. Where applied loads are eventually distributed through a single member within an assembly, the failure of which would result in loss of structural integrity of the component — 99 percent probability with 95 percent confidence.

b. For redundant structure, in which the failure of individual elements would result in applied loads being safely distributed to other load carrying members — 90 percent probability with 95 percent confidence.

c. The effects of temperature on allowable stresses used for design in an essential component or structure must be considered where thermal effects are significant under normal operating conditions.

d. The design of the structure must minimize the probability of catastrophic fatigue failure, particularly at points of stress concentration.

e. Design values greater than the guaranteed minimums required by this paragraph 9 may be used where only guaranteed minimum values are normally allowed; that is, if a "premium selection" of the material is made, in which, a specimen of each item is tested before use, to determine if the actual strength properties of that particular item are equal or exceed those used in design.

109. FABRICATION METHODS. When a fabrication process is used that requires close control, employ methods which consistently produce sound structure and is compatible with the aircraft structure.

a. The methods of fabrication used must consistently produce sound structures. If a fabrication process (such as gluing, spot welding, or heat-treating) requires close control to reach this objective, the process must be performed under an approved process specification.

b. Each new aircraft fabrication method must be substantiated by a test program.

110. FASTENERS. Use hardware conforming to an accepted standard such as AN, NAS, TSO, or MIL-SPEC. Attach equipment in such a way that prevents loosening in service due to vibration.

a. Each removable fastener must incorporate two retaining devices: if one retaining device fails, then the loss of such a fastener would preclude continued safe flight and landing. Alternative to a: Each removable fastener must incorporate two retaining devices to preclude continued safe flight and landing, if one retaining device should fail during flight operations.

b. Fasteners and their locking devices must not be adversely affected by the environmental conditions associated with the particular installation.

c. No self-locking nut may be used on any bolt subject to rotation in operation, unless a non-friction locking device is used in addition to the self-locking device.

111. PROTECTION OF STRUCTURE. Provide protection against deterioration or loss of strength due to corrosion, abrasion, electrolytic action, or other causes. Each part of the structure must:

a. Be suitably protected against deterioration or loss of strength in service due to any cause, including:

(1) Weathering,

(2) Corrosion, and

(3) Abrasion.

b. Have adequate provisions for ventilation and drainage.

112. ACCESSIBILITY. Provide adequate provisions to permit close examination of equipment or adjacent parts of the aircraft that regularly require inspection, adjustment, lubrication, etc. For each part that requires maintenance, inspection, or other servicing, appropriate means must be incorporated into the aircraft design to allow such servicing to be accomplished.

113. AFFECTS ON WEIGHT AND BALANCE. Assure that the altered aircraft can be operated within the weight and center of gravity ranges listed in the FAA Type Certificate (TC), Data Sheet, or Aircraft Listing. Determine that the altered aircraft will not exceed maximum gross weight. (If applicable, correct the loading schedule to reflect the current loading procedure.) Consult Advisory Circular 43.13-1B, Acceptable Methods, Techniques, and Practices-Aircraft Inspection and Repair, for Weight and Balance Computation Procedures.

114. AFFECTS ON SAFE OPERATION. Install equipment in a manner that will not interfere with or adversely affect the safe operation of the aircraft (controls, navigation equipment operation, etc.).

a. The factor of safety prescribed in 14 CFR section 23.303 must be multiplied by the highest pertinent special factors of safety prescribed in 14 CFR sections 23.621 through 23.625 for each part of the structure where strength is:

(1) Uncertain,

(2) Likely to deteriorate in service before normal replacement, or

(3) Subject to appreciable variability because of uncertainties in manufacturing processes or inspection methods.

b. Unless otherwise provided, a factor of safety of 1.5 must be used.

115. CONTROLS AND INDICATORS. Locate and identify equipment controls and indicators so they can be operated and read from the appropriate crewmember position.

116. PLACARDING. Label equipment requiring identification and, if necessary, placard operational instructions. Amend weight and balance information as required.

117. THRU 199. RESERVED

FIGURE 1-4. HYPOTHETICAL OF DETERMINING STATIC TEST LOADS

UTILITY CATEGORY AIRCRAFT (14 CFR PART 23)							
Type of load	Direction	LOAD FACTORS					
		1 Limit	2 X Safety	3 = Ultimate	4 X Special	5 Static = Test	6 Critical Static Test
A. Maneuvering	Fwd	(None)		---------------	--------------	-------------	9.3g
	Down	6.2g	1.5	9.30g	--------------	9.3g	5.7g
	Side	(None)	---------------	---------------	--------------	-------------	
	Up	-3.8g	1.5	-5.7g	--------------	-5.7g	
	Aft	1.0g	1.5	1. 5g	--------------	1.5g	
B. Gust (=30 FPS @ KVc)	Fwd	(None)	---------------	---------------	--------------	-------------	*9.6g
	Down	6.0g	1.5	9.0g	--------------	9.0g	2.4g
*For locations aft of fuselage Sta. 73.85.	Down*	6.4g	1.5	9.6g	--------------	9.g	
	Side	1.6g	1.5	2.4g	--------------	2.4g	
	Up	-2.8g	1.5	-4.2g	--------------	-4.2g	
	Aft	(None)	--------------	---------------	--------------	-------------	
C. Ground	Fwd	6.6g	1.5	9.9g	--------------	9.9g	9.9g
	Down	4.0g	1.5	6.0g	--------------	6.0g	
D. Ultimate Inertia Forces for Emergency Landing Condition (Section 23.561).	Fwd	Already Prescribed as Ultimate		9.0g	--------------	-------------	**4.5g
	Fwd.**			4.5g	--------------	-------------	
	Down			(None)	--------------	-------------	
**For Separate cargo compartments.	Side	Already Prescribed as Ultimate		1.5g	--------------	1.5g	
	Up			-3.0g	--------------	-3.0g	
	Aft			(None)	--------------	-------------	

FIGURE 1-4. HYPOTHETICAL OF DETERMINING STATIC TEST LOADS – CONTINUED

UTILITY CATEGORY AIRCRAFT (14 CFR PART 23)							
Type of load	Direction	LOAD FACTORS					
		1 Limit	2 X Safety	3 = Ultimate	4 X Special	5 Static = Test	6 Critical Static Test
E. Ultimate Inertia Forces for Emergency Landing Condition For Seat, Litter, & Berth Attachment to Aircraft Structure (Section 23.785).	Fwd	Already Prescribed as Ultimate		9.0g	1. 33	12.0g	12.0g
	Down			(None)	--------------	--------------	
	Side			1. 5g	1.33	2.0g	
	Up			-3.0g	1.33	-4.0g	
	Aft			(None)	--------------	--------------	

* Asterisks denote special load conditions for the situation shown.

CHAPTER 2. COMMUNICATION, NAVIGATION, AND EMERGENCY LOCATOR TRANSMITTER SYSTEM INSTALLATIONS

200. PURPOSE. The purpose of this chapter is to describe installation considerations and requirements for basic, stand alone, installations of communication, navigation, and emergency locator transmitter (ELT) equipment.

NOTE: Stand alone installations do not depend on other systems or complex interfaces to function.

201. HAZARDS/WARNINGS.

a. When installing these systems follow the aircraft and equipment manufacturers' instructions as appropriate. Ensure that equipment and systems function properly and perform their intended function(s).

b. Alterations of aircraft that are performed to accommodate the installation of radio equipment must be evaluated for their impact on aircraft design and operation. Refer to Advisory Circular (AC) 23.1309-1, Equipment, Systems, and Installations in Part 23 Airplanes, (as amended) for additional information concerning the evaluation for equipment, systems, and installations in Title 14 of the Code of Federal Regulations (14 CFR) part 23 airplanes.

c. Frequently an alteration to accommodate the installation of radio equipment will have little impact on the design or operation of an aircraft; however, all potential elements of impact must be considered. One approach is to evaluate each element independently.

d. For example: The weight of a component or total weight of a system may exceed the design weight or balance of an aircraft. The structural load (and required allowances) presented at a component attach point may exceed the design structural limits at that location. Radio frequency energy radiated from a system may negatively affect another existing system (e.g., accuracy of the wet compass).

e. When evaluating elements of impact consider the before and after states. No further analysis of weight and balance may be required if an object of similar size and weight was previously installed in a location. If mounting attach points were previously substantiated to support a specific load and the same load or less is being installed, the previous analysis may be referenced.

NOTE: Data that is referenced must be available and reviewed.

f. When structures must be fabricated or reinforced, the standard practices approved for repairs if applicable may be employed. AC 43.13-1, (as amended) Acceptable Methods, Techniques, and Practices—Aircraft Inspection and Repair, may provide structural design data for fabrication of mountings and attachments.

g. Ensure that the capacity of the aircraft's charging system is not exceeded, including any required additional allowances.

h. Care should be taken to ensure that cables or wires will not interfere with the aircraft's flight, engine, or propeller controls.

i. When removing older radios/wiring/power supplies and installing newer solid state components weigh the old equipment and perform a new weight and balance calculation. This is important since differences in the location and weight of equipment will shift the center of gravity.

j. Refer to AC 43.13-1, chapter 10 (as amended) and FAA-H-8083-1 for additional information concerning determination of weight and balance.

202. CONSIDERATIONS WHEN INSTALLING AVIONICS EQUIPMENT. When installing radio equipment, use areas or locations designated by the airframe manufacturer and use factory supplied brackets or racks. Follow the aircraft manufacturer's installation instructions. When this information is not available, use locations in the aircraft of known load carrying capabilities. Baggage compartments and cabins or cockpit floors are good mounting platforms provided the floor attachments meet the strength requirements. Another method is to fabricate support racks, brackets, or shelves and attach them to the aircraft structure to provide a mounting that will withstand the inertia forces stipulated in chapter 1.

a. Ensure the Following.

(1) There is appropriate air circulation to ensure proper cooling and dissipation of any heat generated or present. Consider the flammability characteristics of all associated elements.

(2) There are appropriate clearances to prevent mechanical damage to other parts of the aircraft or from other parts of the aircraft.

(3) There is protection provided to the component or article from any fluids or fumes that may be expected and that the component or article will not cause any fluids or fumes to be present that may result in damage to the aircraft or its occupants.

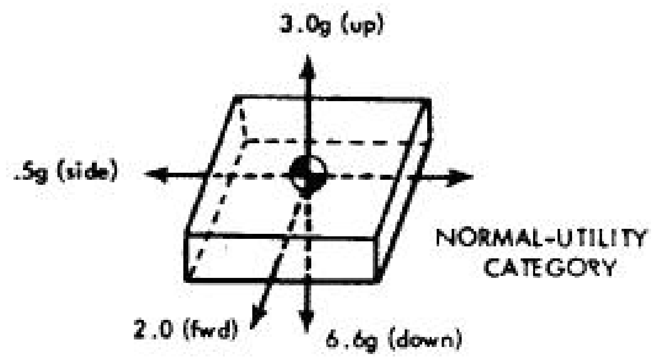
(4) That any interference, environmental or operational, to an aircraft or any system of the aircraft is identified and minimized as to not affect airworthiness of the aircraft.

(5) That flight characteristics of the aircraft are not altered unless appropriately identified and the changes are within the certified design limits.

b. Structural Consideration.

(1) Structural requirements of a mounting must be considered (see chapter 1).

FIGURE 2-1. FORCE DIAGRAM NORMAL-UTILITY CATEGORY



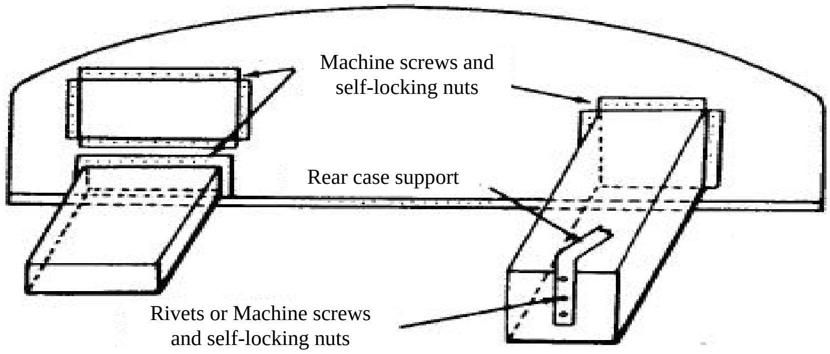
(2) Alterations that include making additional cutouts or enlargements of existing panel cutouts must be evaluated to maintain structural integrity. Some aircraft instrument panels are load bearing structures.

(3) Loads must be determined to be within the structural design limits of the supports.

(4) Instrument panels as well as other panels throughout the aircraft may be structural or nonstructural in design. Structural loads must be adequately transferred to primary airframe members.

(5) Methods and practices described elsewhere in this AC and in AC 43.13-1 (as amended) may be employed for the fabrication of attachments and structure.

FIGURE 2-2. TYPICAL FABRICATED PANEL MOUNTING



(6) Existing structures may be reinforced or strengthened using methods described in AC 43.13-1 (as amended).

203. INSTRUMENT PANEL MOUNTING. This paragraph is supplemented by AC 43.13-1, chapter 2, (as amended) and is applicable to the installation of radio units in instrument panels.

FIGURE 2-3. TYPICAL LAYOUTS

Typical layout older aircraft

Typical layout newer aircraft

a. Stationary Instrument Panels—Nonstructural and Structural. The stationary instrument panel in some aircraft is the primary structure. Prior to making any additional "cutouts" or enlargement of an existing "cutout," determine if the panel is the primary structure. If the panel is structural, make additional "cutouts" or the enlargement of existing "cutouts" in accordance with the aircraft manufacturers' instructions, or substantiate the structural integrity of the altered panel in a manner acceptable to the Administrator. Radius all corners and remove all burrs from "cutout" and drilled holes.

b. Added Equipment Stationary Instrument Panel. When radio equipment is to be installed in a stationary panel already supporting instruments, glove compartments, etc., determine the capability of the panel to support the total load.

c. Case Support. To minimize the load on a stationary instrument panel, whenever practicable, install a support between the rear (or side) surface of the radio case and a nearby structural member of the aircraft (Figure 2-2).

d. Added Equipment—Shock-Mounted Panels. When installing radio equipment designed for use in shock-mounted panels, total accumulated weight of equipment installed must not exceed the weight carrying capabilities of the shock mounts. Determine that the structure to which the shock mounts are connected is satisfactory for supporting the added weight.

e. Existing Factory Fasteners. When possible, use existing plate nuts and machine screws provided by the aircraft manufacturer for attachment of the radio case or rack. If additional fastening is required, use machine screws and elastic stop nuts (preferably plate nuts).

f. Magnetic Direction Indicator. As a function of the radio installation, determine if it is necessary to swing the compass. Install a suitable placard which indicates the compass error with the radio(s) on and off. When inverters are installed, determine what effect their operation has on the magnetic compass. Maximum acceptable deviation in level flight is 10 degrees on any heading. The following is an example of a typical compass calibration card (refer to AC 43.13-1B, chapter 12 on how to swing a compass).

TABLE 2-1. TYPICAL COMPASS CALIBRATION CARD

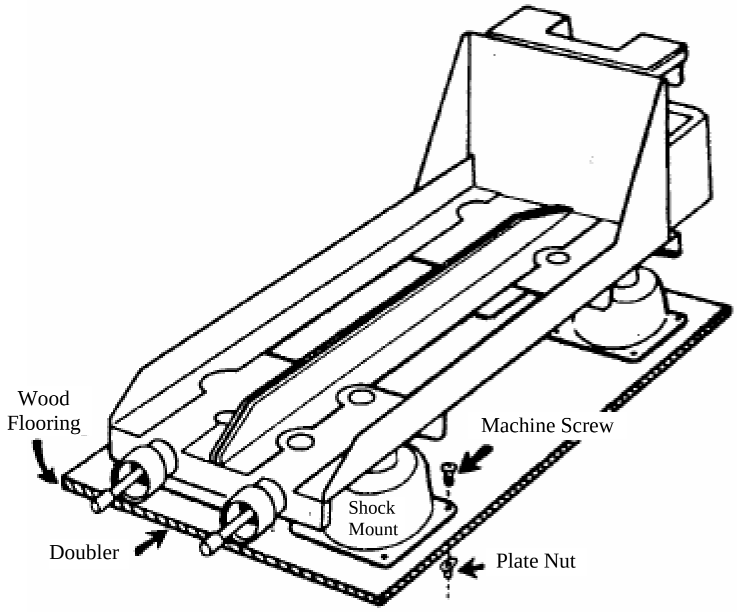
FOR	N	30	60	E	120	150
Radio On Steer	4°	35°	63°	93°	123°	154°
Radio Off Steer	358°	27°	58°	88°	118°	148°
FOR	S	210	240	W	300	330
Radio On Steer	183°	214°	224°	274°	304°	337°
Radio Off Steer	178°	208°	238°	268°	293°	327°

204. OTHER MOUNTING AREAS. The following are acceptable methods for installing radio equipment at other than instrument panel locations.

a. Shock-mounted Units.

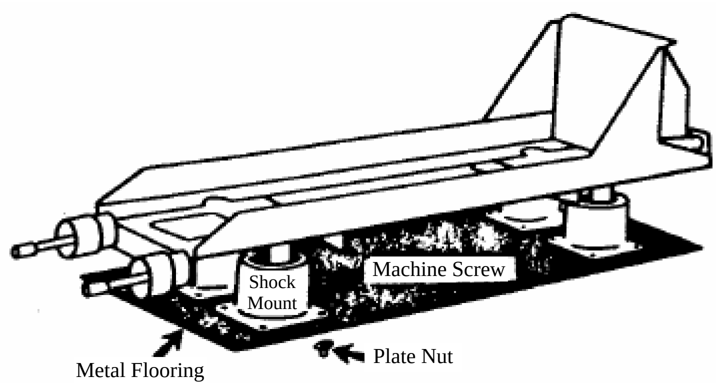
(1) Wood or Composition Flooring. Secure the shock-mounted base assembly (suitable to radio unit) directly to the floor using machine screws. Add a doubler to the bottom of the floor thereby sandwiching the composition floor between each shock-mount foot and the doubler. Subsequent removal and reinstallation of the shock-mount foot will be facilitated if plate nuts are secured to the doubler. Where practicable, use small retaining screws to keep the doubler in position. Install a ground strap between the radio rack and metal structure of the aircraft.

FIGURE 2-4. TYPICAL SHOCK-MOUNTED BASE



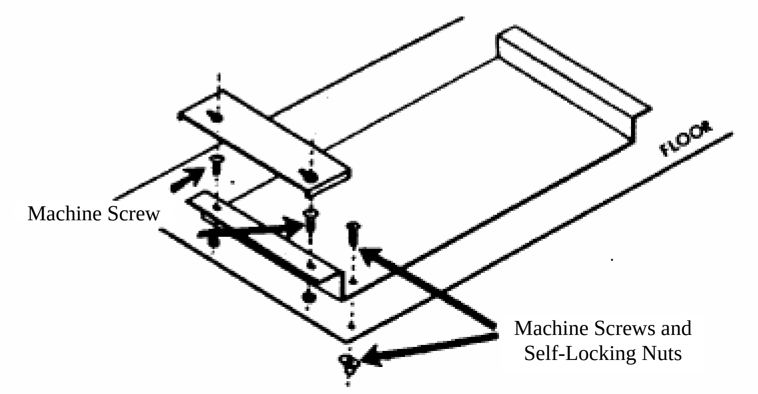
(2) Metal Flooring. Secure the shock-mounted base assembly directly to the floor using machine screws, washers, and self-locking nuts. Floor area under and around the radio mounting bases may require installation of doublers or other reinforcement to prevent flexing and cracking. Installation of plate nuts on the floor or doubler will facilitate removal and installation of the shock mounts. Install a ground strap between the shock mount foot and the radio rack.

FIGURE 2-5. TYPICAL SHOCK MOUNTED BASE



b. Rigid-Mounted Unit Base. Secure radio mounting base plate(s) to the floor (wood, composite, or metal) using machine screws as shown in Figure 2-4. Use a reinforcing plate or large area washers or equivalent under wood or composite flooring. When the mounting base is secured to wood or composite material, install a ground strap between the base and aircraft metal structure.

FIGURE 2-6. TYPICAL RIGID BASE PLATE MOUNT



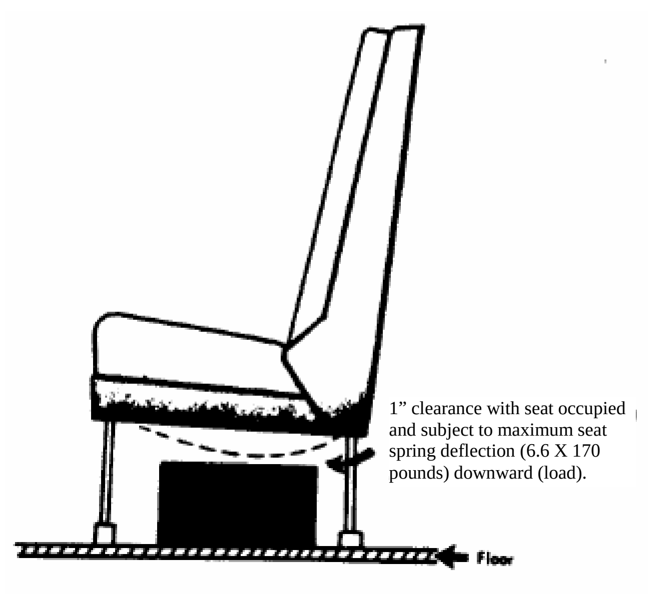
205. FABRICATION OF SUPPORTING BRACKETS FOR ATTACHMENT TO STRUCTURE OTHER THAN FLOORING.

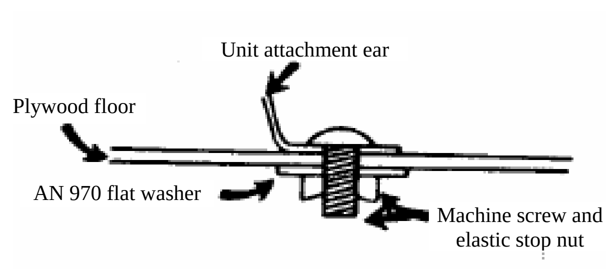
a. Typical supporting brackets usually consist of a shelf or platform upon which the radio unit mounting base assembly can be installed in the same manner as described in applicable Paragraph 3.

b. Fabricate bracket in accordance with good aircraft design, layout, assembly practices, and workmanship to obtain results compatible with the airframe structure. Generally, the thickness of bracket material will depend on the size or area of the platform and load it must sustain in accordance with provisions set forth in Chapter 1 of this handbook.

c. Use a rivet size and pattern compatible with the aircraft structure to provide the strength needed to assure support of the loads imposed under all flight and landing conditions.

FIGURE 2-6. TYPICAL UNDERSEAT INSTALLATION





NOTE: To increase the strength of floor attachment points, metal reinforcement may be installed as needed.

206. REINFORCEMENT OF SUPPORTING STRUCTURE.

a. Attach equipment to the supporting structure of the aircraft so that its supported load will be transmitted to aircraft structural members. If direct attachment to the existing structure (bulkheads, horizontal stringers, etc.) is not feasible, add the necessary stringers, doublers, bulkhead flange reinforcements, etc., to provide adequate support and assure load transfer to the primary structure. When attaching to the existing structure ensure that the attachment does not weaken the structure.

b. Placard. Fasten onto the shelf or bracket a permanent placard (as the example below) stating the design load which the installed structure is determined capable of supporting.

"Shelf load not to exceed

__________lbs."

FIGURE 2-7. TYPICAL REMOTE UNIT MOUNTING BASE-VERTICAL OR HORIZONTAL

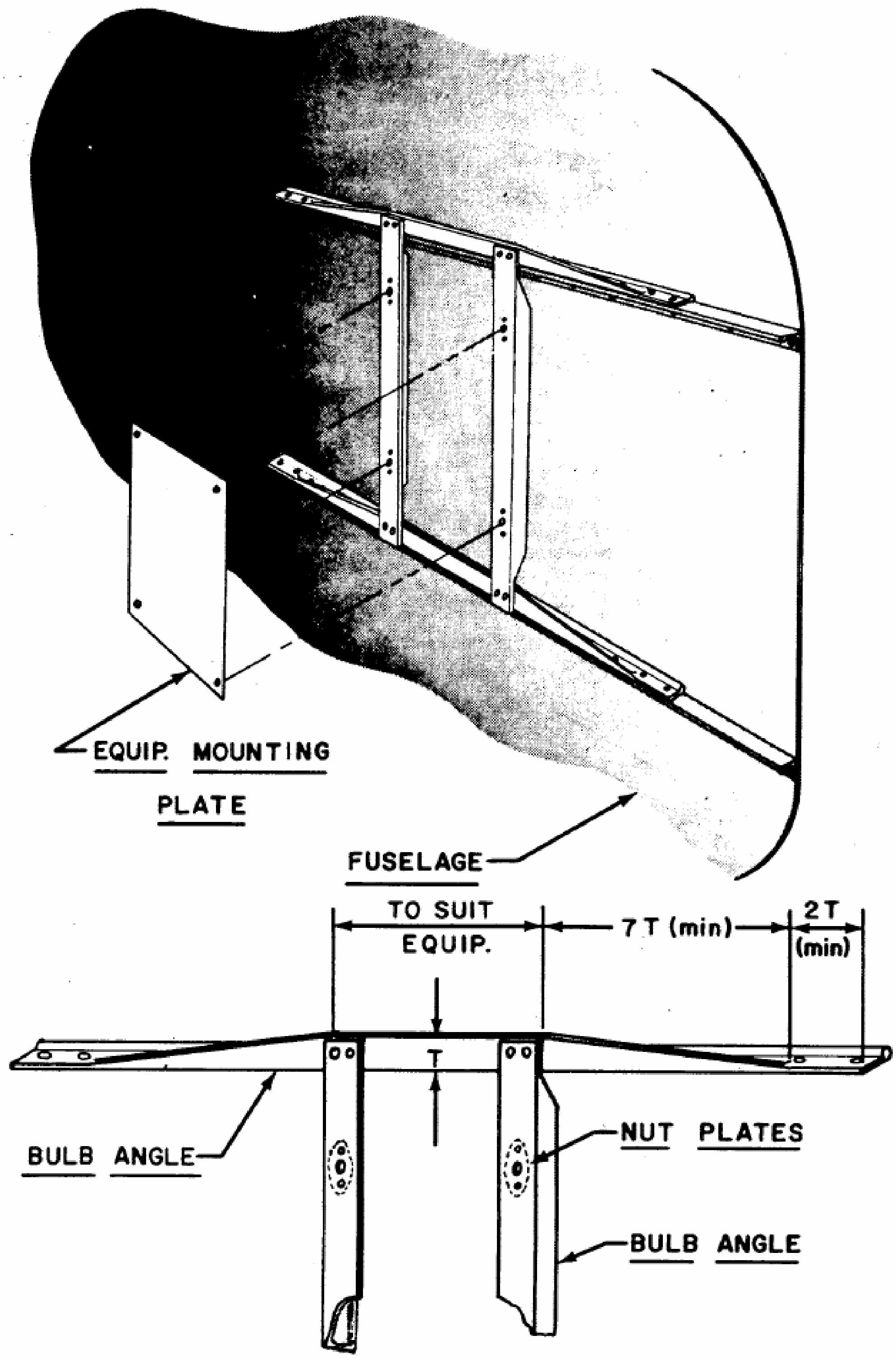
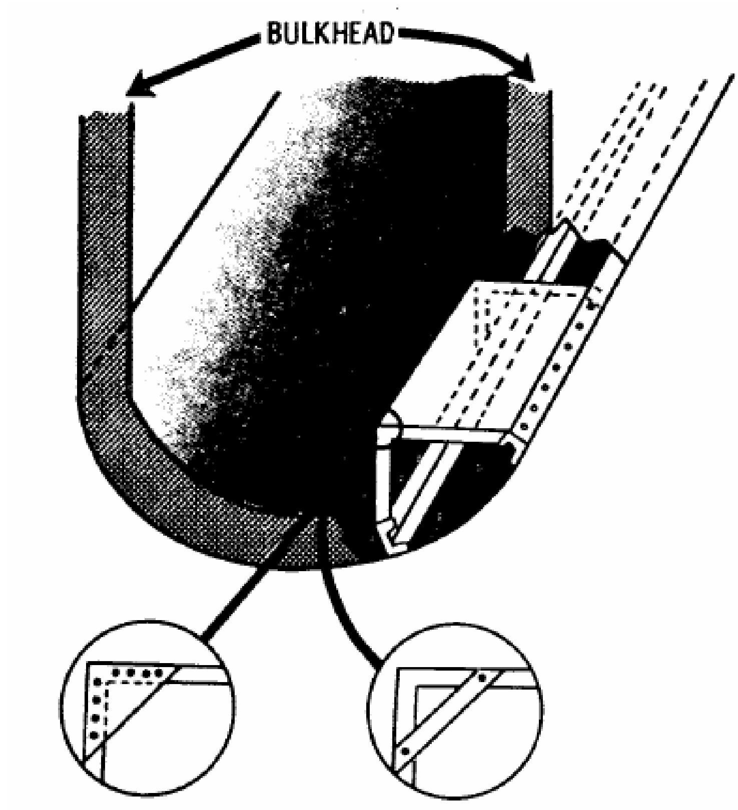


FIGURE 2-8. TYPICAL SHELF INSTALLATION

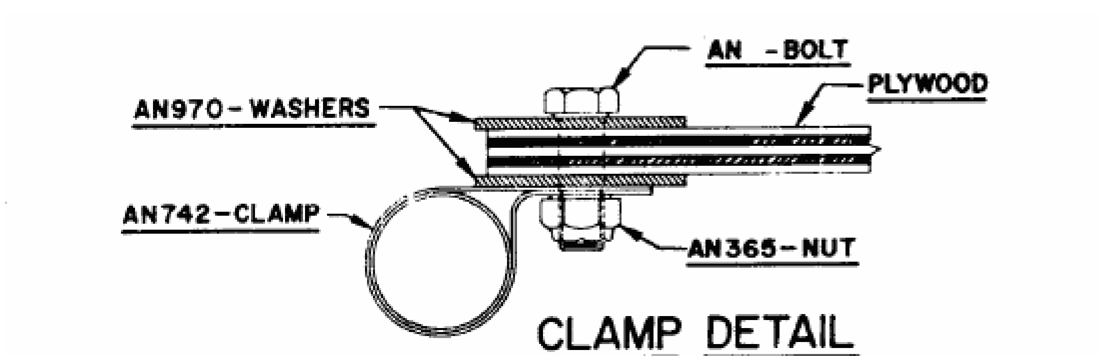


NOTE: Use standard aircraft practices and procedures for fabrication and attachment of the shelf. Reinforce fore and aft corners with gussets or bulb angle.

NOTE: Fabricate a platform using 2017T4(17ST) or equivalent. Apply standard aircraft practices for fabrication and installation.

NOTE: The equipment manufacturer mounting bases that meet load requirements and can be utilized are acceptable.

FIGURE 2-10. TYPICAL ATTACHMENT OF SUPPORT STRUCTURE TO TUBULAR FRAME



207. ELECTRICAL REQUIREMENTS.

a. Installation of an electrical system or component into an aircraft requires consideration of the electrical load, the appropriate power distribution circuit, and available power capacity. The specific requirements for a system are a product of its electrical characteristics and the criticality of its application for use in the operation of the aircraft.

b. The total energy available to power electrical systems is referred to as the aircraft's capacity. It includes available stored power and generated power. This will vary depending on phase of flight or type of operation.

c. The critical distribution circuit of an aircraft is designed to transfer power from source to a system determined critical to the operation or function of the aircraft. This circuit is required to include additional capacity and circuit protection as required by applicable regulations. Distribution systems should be designed to facilitate load-shedding procedures. Power distribution system design includes the following concerns. Reference AC 43.13-1, chapter 11 (as amended) for:

(1) Circuit Protection.

(2) Wire selection.

(3) Connectors, switches, and termination devices.

(4) Wire routing.

(5) Wiring/cable support.

(6) Identification.

d. Installation of Wiring.

(1) Use a type and design satisfactory for the purpose intended.

(2) Install in a manner suitably protected from fuel, oil, water, other detrimental substances, and abrasion damages.

e. Power Sources.

(1) Connect radio electrical systems to the aircraft electrical system at the power source protective device, a terminal strip, or use a plug and receptacle connection.

(2) Radio electrical systems must function properly whether connected in combination or independent.

f. Protective Devices.

(1) Incorporate a "trip free" re-settable type circuit breaker or a fuse in the power supply from the bus. Mount in a manner accessible to a crewmember during flight for circuit breaker resetting or fuse replacement.

(2) Select circuit breakers or fuses that will provide adequate protection against overloading of the radio system circuits.

(3) Connect all leads in such a manner that the master switch of the aircraft will interrupt the circuit when the master switch is opened, unless the equipment is intended to be powered when the master switch is open.

(4) Radio system controls are to provide independent operation of each system installed and be clearly labeled to identify their function relative to the unit of equipment they operate.

g. Wire Bundle Separation from Flammable Fluid Lines.

(1) Physically separate radio electric wire bundles from lines or equipment containing oil, fuel, hydraulic fluid, alcohol, or oxygen.

(2) Mount radio electrical wire bundles above flammable fluids lines and securely clamp to the structure. (In no case must radio electrical wire bundles be clamped to lines containing flammable fluids.)

h. Cable Attachment to Shock-Mounted Units. Route and support electrical wire bundles and mechanical cables in a manner that will allow normal motion of equipment without strain or damage to the wire bundles or mechanical cables.

i. Radio Bonding. It is advisable to electrically bond radio equipment to the aircraft in order to provide a low impedance ground and to minimize radio interference from static electrical charges. When electrical bonding is used, observe the following:

(1) Keep bonding jumpers as short as possible.

(2) Prepare bonded surfaces for best contact (resistance of connections should not exceed .003 ohm).

(3) Avoid use of solder to attach bonding jumpers. Clamps and screws are preferred.

(4) For bonding aluminum alloy, use aluminum alloy or tinned or cadmium-plated copper jumpers. Use brass or bronze jumpers on steel parts.

(5) When contact between dissimilar metals cannot be avoided, put a protective coating over the finished connection to minimize corrosion.

208. ELECTRICAL LOAD ANALYSIS PROCEDURE.

a. Available Power Supply. To preclude overloading the electric power system of the aircraft when additional equipment is added, perform an electrical load analysis to determine whether the available power is adequate. Radio equipment must operate satisfactorily throughout the voltage range of the aircraft electrical system under taxi, takeoff, slow cruise, normal cruise, and landing operating conditions. If night and instrument flight is contemplated, compute the electrical load analysis for the above flight regimes under the most adverse operating conditions.

b. One method for the analysis may be found in the ASTM International, Standard Guide for Aircraft Electrical Load and Power Source Capacity Analysis, F 2490-05.

c. Applicable elements of a previously performed electrical load analysis if available may be reused.

209. ELECTROMAGNETIC COMPATIBILITY.

a. Electromagnetic Interference (EMI) may disrupt the performance of systems and has varying degrees of consequence. These consequences may be identified as: none, minor, major, hazardous, or catastrophic. The purpose of electromagnetic compatibility (EMC) analysis and testing is to assure that equipment does not cause interference with any existing aircraft system function, and that existing systems do not cause any interference with the new equipment.

TABLE 2-2. RELATIONSHIP PROBABILITIES, SEVERITY OF FAILURE CONDITIONS

Classification of Failure Conditions	No Safety Effect	<--Minor-->	<--Major-->	<--Hazardous-->	<--Catastrophic-->
Effect on Airplane	No effect on operational capabilities or safety	Slight reduction in functional capabilities or safety margins	Significant reduction in functional capabilities or safety margins	Large reduction in functional capabilities or safety margins	Normally with hull loss
Effect on Occupants	Inconvenience for passengers	Physical discomfort for passengers	Physical distress to passengers, possibly including injuries	Serious or fatal injury to an occupant	Multiple fatalities
Effect on Flightcrew	No effect on flightcrew	Slight increase in workload or use of emergency procedures	Physical discomfort or a significant increase in workload	Physical distress or excessive workload impairs ability to perform tasks	Fatal Injury or incapacitation

b. When EMC characteristics are known, the need for and extent of EMC testing can be determined by a review of those characteristics. Knowing specific target frequencies, EMC testing can focus on the aircraft systems (and even those susceptible frequencies in the case of tunable systems) likely to be affected by interference. Where sensitive systems or potentially strong sources of EMI are involved, more intensive evaluation will be required.

TABLE 2-3. COMMON USED RADIO FREQUENCIES ON AIRCRAFT

Range	Hz	Mode	Function
190-1750	kHz	Rx	ADF Navigation
2-30	MHz	Tx/Rx	HF Communications
75	MHz	Rx	Marker Beacon Receiver
108-112	MHz	Rx	ILS Localizer Receiver
108-118	MHz	Rx	VHF Omnirange (VOR) Receiver
118-137	MHz	Tx/Rx	VHF Communications
243	MHz	Tx	Emergency Locator Xmtr (Satellite)
328.6-335.4	MHz	Rx	ILS Glide Slope Receiver
406.3	MHz	Tx	Emergency Locator Transmitter
960-1215	MHz	Tx/Rx	DME System
1027-1033	MHz	Tx/Rx	Transponder & TCAS Systems
1087-1093	MHz	Tx/Rx	Transponder & TCAS Systems
1575.42	MHz	Rx	GPS Satellite Navigation

c. On-aircraft EMC tests for systems or equipment should be conducted. If lab test data is available it should be used to guide the planning of these tests. If lab test analysis is not available a more comprehensive EMC testing on the aircraft systems and equipment typically needs to be performed.

(1) The aircraft should not be close to large reflecting surfaces such as buildings or other aircraft. Use of ground power is not recommended, as ground power units are not routinely checked for output quality.

(2) All normally closed circuit breakers should be closed and power should be supplied to all normally powered AC and DC distribution busses during testing.

(3) The aircraft should be in flight configuration. Doors and hatches that might be in any interfering signal's path should be in the position they would normally be in during flight.

(4) Aircraft systems being tested should be operated and monitored for indication of interference. (It is essential that systems determined critical for the operation of the aircraft are tested.)

TABLE 2-4. AIRCRAFT ELECTRICAL AND ELECTRONIC SYSTEMS

The following systems, if installed, should be included in the aircraft EMC test plan.

System
ADF
Air Data Systems
Altitude Alert System
ATC Transponder
Audio Distribution System
Autopilot/Flight Guidance System
Compass/Directional Gyro Systems
DME
Electronic Flight Control System
Electronic Flight Instrument System (EFIS)
Global Positioning System (GPS)
Marker Receiver System
VHF Communications
VOR/LOC/GS
Newly Installed Electrical/Electronic System

210. FUNCTIONAL HAZARD ASSESSMENT.

a. Many older aircraft designs did not provide for all weather operations or potential increases in the pilot's reliance on installed systems and equipment. Requirements to assure design safeguards against hazards have developed as technologies have become available and pilots have increased their reliance on installed systems and equipment. A fundamental analysis to assure design safeguards is accomplished through a functional hazard assessment.

b. Alterations that involve systems or equipment that perform critical functions or that include complex designs that have a high degree of integration, use of new technology, or novel applications of conventional technology, must be assessed to determine the severity of failure conditions. Complexity in itself does not drive the need to perform a system safety analysis but the effect of a failure does. Comparison with similar, previously approved systems is sometimes helpful.

c. Evaluate the system to determine if it is essential or not essential to safe operation. For aircraft of 6,000 pounds or less maximum weight, refer to the regulations incorporated by reference in the type certificate, unless the Administrator has found that the change is significant in an area.

d. Determine if the equipment has any unacceptable, adverse affect when operated.

e. Determine if the operation of the installed equipment has an adverse affect on equipment not essential to safe operation, and if a means exists to inform the pilot of the effect.

f. Determine if a failure or malfunction of the installed equipment could not result in unacceptable hazards.

g. Design requirements and methodology of hazard resolution differing upon application and type of aircraft (i.e., single-engine, multiengine, commuter use).

(1) Operation of equipment that has an adverse effect on other equipment essential to safe operation of the aircraft is unacceptable.

(2) Operation of equipment that has an adverse effect on other equipment that is not essential to safe operation of the aircraft may be acceptable if there is a means to inform the pilot of the effect.

(3) If a probable failure or malfunction will result in a hazard in a multiengine aircraft it is unacceptable.

(4) If a probable failure or malfunction will result in a hazard in a single-engine aircraft its impact must be minimized.

211. THRU 299. RESERVED

CHAPTER 3. ANTENNA INSTALLATION

300. PURPOSE. The purpose of this chapter is to describe antenna installation methods and practices. An antenna that is installed on an aircraft must function properly and may not adversely affect other systems or equipment.

301. HAZARDS/WARNINGS.

a. Follow antenna manufacturer's instructions and recommendations when they are available and appropriate, and not contrary to the instructions of the aircraft manufacturer.

b. Extension of the landing gear or flaps may impact belly mounted antenna performance.

302. ADDITIONAL REFERENCES. For further information concerning acceptable methods, techniques, and practices concerning alteration involving specific structures refer to the appropriate chapter of Advisory Circular (AC) 43.13-1, Acceptable Methods, Techniques, and Practices―Aircraft Inspection and Repair (as amended).

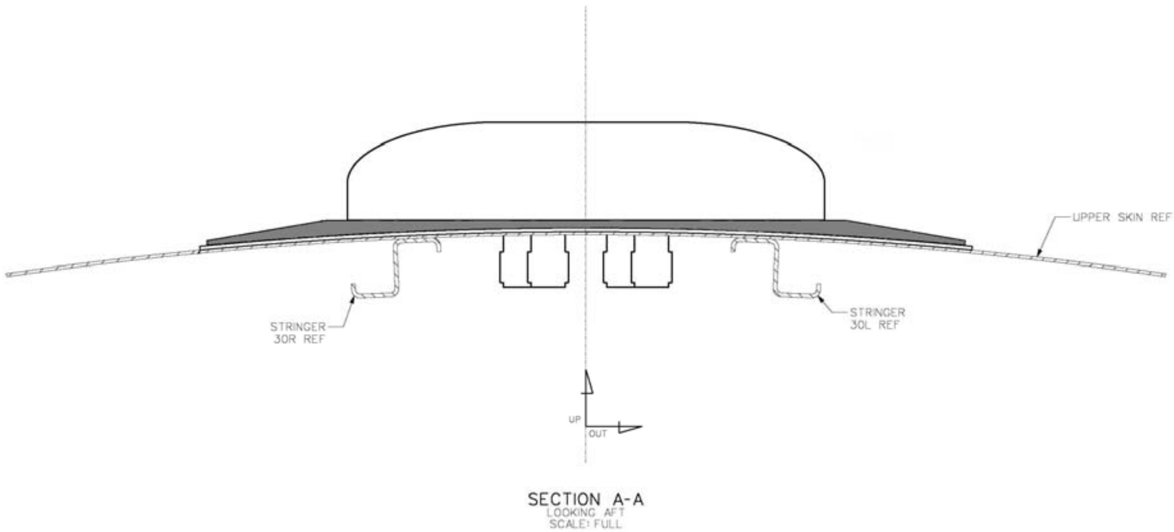
303. STRUCTURAL SUPPORT.

a. The antenna's structural load, plus any required allowances, may not exceed the design capacity of the structure intended to support it. It is important to understand the operational characteristics of the aircraft and consider forces that occur during flight (dynamic loading) as well as those that occur when the aircraft is not in motion (static loading). For example, an aircraft designed without flaps may employ a side slip procedure, to lose altitude, during which the direction of airflow across the fuselage is not in line with the aircraft longitudinal axis. Antenna mountings on these aircraft need to be designed and evaluated for the direction of airflow that occurs during such an operation.

b. Whenever possible, an antenna should be mounted to a flat surface. Minor aircraft skin curvature can be accommodated with the use of an appropriate gasket but if gaps over .020" appear between the base plate and mounting surface, use of a mounting saddle is recommended.

c. Since antenna systems typically require a ground plane (this may be a conductive surface that the antenna mounts to) any separation of an antenna from its ground plane may impact performance. Contact the manufacturer for recommendations if a gasket or mounting saddle is needed.

FIGURE 3-1. ANTENNA MOUNT WITH SADDLE



d. Mounting screws must never be over torqued in an attempt to distort aircraft structure to reduce gaps between the antenna base plate and aircraft-mounting surface.

e. Consider the factors of flutter, vibration characteristics, and drag load. The approximate drag load an antenna develops may be determined by the formula:

$$D=.000327\ AV^2$$

(The formula includes a 90 percent reduction factor for the streamline shape of the antenna.)
D is the drag load on the antenna in lbs.
A is the frontal area of the antenna in sq. ft.
V is the V_{NE} of the aircraft in m.p.h.

Example: Antenna manufacturer specification frontal area = .135 sq. ft. and V_{NE} of aircraft is 250 m.p.h.

$$D=.000327 \times .135 \times (250)^2$$
$$=.000327 \times .135 \times 62{,}500$$
$$= 2.75 \text{ lbs}$$

f. The above formula may be adapted to determine side load forces by substitution of the apparent frontal area value for A, when the aircraft motion and antenna orientation are not the same.

304. PHYSICAL INTERFERENCE.

a. Antennas should be located where it will not interfere with the operation of the aircraft or other aircraft systems.

b. Antennas should be located so that they don’t obstruct or limit airflow to areas of the airframe that require airflow. Care should be exercised that an antenna is not located where it will be damaged by heat from engine exhaust, fumes from battery vents or fuel/fluid drains.

c. Antennas may accumulate ice that can then depart and damage areas behind them. Special attention should be paid to areas of the airframe near pitot static ports and sensors and near flight controls, since the antenna may alter airflow characteristics.

d. Antennas should be located in such a manner that they are not susceptible to damage from misuse such as near a door where they might be mistaken for or used as a handhold.

305. ANTENNA SELECTION. The selection of an appropriate antenna will include consideration of system requirements and aircraft characteristics. The size and shape of an antenna varies with frequency, power rating, and maximum design speed of the antenna. See Figures 3-7 through 3-19 for pictures of typical antennas.

306. ANTENNA LOCATION.

a. In general, antenna locations on an aircraft, which provide unobstructed line-of-sight views of the transmitted or received signals, are preferred. Objects located in the path of a signal may cause a blanking of antenna coverage and impact the performance of the system.

NOTE: Global Positioning Service (GPS) antennas will not receive a signal if a line-of-sight view of a satellite is not available. Do not mount a GPS antenna on the underside of an aircraft.

b. Acceptable and unacceptable spacing between an antenna and an obstacle or the permissible interval between antennas is dependent upon operating frequency and system characteristics. When in doubt contact the antenna manufacturer or system designer for further information.

c. Antennas should be separated as far as possible from interference sources (other radiating antennas, ignition noise sources, etc.). When known interference sources are present it may be advisable to temporarily position and antenna and check a location for suitability prior to mounting the antenna.

d. VHF Com 1 should be mounted on the top of the aircraft since this will provide the best unobstructed location. VHF Com 2 can also be mounted on the top, provided there is at least 1/2 wavelength (of the antenna operating frequency) distance available between antennas.

e. If Com 2 is mounted under the aircraft, a bent whip may be required to provide ground clearance. Bent whips may not provide the best performance because of proximity to the aircraft skin. Signal reflection and obstruction is more of a problem with such locations. Extension of the landing gear or flaps may also impact belly mounted antenna performance.

f. Antennas need to "see" with a direct line-of-site to the source. Antenna patterns can be disrupted by landing gear or vertical stabilizers as examples. When mounting antennas, try to locate them in areas where line-of-sight view is not obstructed.

g. As a rule of thumb, maintain 36 inches as a minimum distance between antennas. Refer to manufacturer's installation guidelines for specific system limitations and requirements.

h. Antennas should be located such that cable runs between antenna and equipment are as short as practical. Signal loss of a cable is dependent upon cable design, length, and frequency. In some cases, cables must be specific lengths to provide a required capacitance, attenuation, or signal transmission time. VOR/LOC/GS blade or towel bar type antennas require the cables to their coupling assembly to be the exact same length to maintain a phase relationship.

307. ANTENNA BONDING.

a. The electrical bonding of the antenna to the aircraft surface is extremely important. The conductive skin of an aircraft is an electrical part of the antenna system. If an antenna is not properly bonded to the aircraft, its pattern may be distorted and nulls in coverage may appear.

b. The electrical bonding of the antennas to the aircraft skin of a metal aircraft is best accomplished by direct metal-to-metal contact of the antenna base to the skin. A resistance of no more than .003 ohms between the antenna base plate and skin should be achieved.

- To achieve this electrical bonding, the aircraft paint in the mounting area will need to be removed and the surface alodined to protect aluminum against corrosion

c. An alternate method for providing electrical bonding to metal aircraft skin is through the antenna mounting screws, which attach to a backing plate inside the aircraft, making electrical contact with the backside of the skin. To ensure good contact remove any interior paint in the area where the backing plate is placed and coat this area with alodine to minimize corrosion.

d. Composite or fabric covered aircraft that do not provide a conductive mounting surface generally require fabrication of a conductive surface (ground plane) and bonding through the mounting screws.

e. Antenna performance can be severely degraded from corrosion caused by moisture accumulation where the antenna electrically bonds to its ground plane. It is advisable to apply RTV around the antenna edges to seal the antenna bond; however, always ensure chemical compatibility before using any sealant.

308. ELECTROMAGNETIC INTERFERENCE (EMI).

a. Since the purpose of an antenna is to either receive or transmit RF energy (or both) it is essential to consider EMI.

b. Antenna mounting positions should be selected that are as far as possible from an EMI source. In special cases it may be possible to employ filters or select an antenna that has been specifically designed to be resistant to an EMI.

c. EMI test procedures may be found in AC 43.13-2B, chapter 2.

309. MECHANICAL INSTALLATION.

a. Mounting Hardware.

(1) Typical antenna installations employ either #8-32 or #10-32 stainless steel mounting screws.

(2) Some designs require a pan head screw, others a counter sunk screw is required.

(3) Mounting screw length will vary based on each particular installation requirement.

b. General Practices.

(1) Refer to installation drawing before drilling holes in aircraft skin to determine proper size and spacing.

NOTE: When replacing antennas it is important to match the original mounting holes. Previous mounting holes that are not reused with appropriate hardware must be repaired and the mounting location returned to its design strength.

(2) Use of a structural backing plate is highly recommended. Backing plates strengthen the immediate point of attachment but if they are not attached to load carrying structure they do not provide structural load support.

(3) Mounting screws should be secured with stainless steel nuts with flat washers and lock washers, or with flat washers and lock nuts to secure the antenna properly.

(4) Sandwich the aircraft skin between the antenna base plate and the internally mounted backing plate. Before securing the antenna to the aircraft make sure that all the cables are connected to the unit and fit through the connector holes in the aircraft.

(5) Gently tighten the mounting hardware so that uniform stress is placed on each side of the antenna. For #8-32 screws DO NOT exceed 20 in•lbs of torque and for #10-32 screws DO NOT exceed 23 in•lbs of torque. Refer to fastener manufacturer's torque guidelines to confirm that these recommended settings do not exceed the chosen fasteners torque limits.

(6) Once the antenna is mounted, any minor gaps between the base plate or gasket and aircraft skin should be filled with RTV silicone adhesive sealant.

(7) Double check that a reading of .003 ohms between the antenna base plate and ground has been achieved.

310. GROUND PLANE REQUIREMENT (WOOD/FABRIC AND COMPOSITE AIRCRAFT).

a. When the antenna is not mounted to a conductive surface capable of providing a required ground plane for operation a ground plane must be fabricated.

b. Most antennas require a ground plane size of approximately 24" by 24". While the rule of thumb is to provide a minimum of 1/4 wavelength of the operation frequency, larger is better and ground plane symmetry is critical. Gaps in antenna coverage or performance may occur if a ground plane is not symmetrical.

TABLE 3-1. WAVELENGTH IN FEET = 984 / FREQUENCY IN MEGAHERTZ (MHZ)

Frequency	Wavelength (ft)	1/4 wavelength (in)
75 MHz	13.1	40
125 MHz	7.9	24
1000 MHz	1	3

c. Wire mesh is the best material to use when a solid plate is not practical. Heavy aluminum foil can also be used. In all cases, electrical continuity from ground plane to airframe ground is essential.

d. Be sure the ground plane is well attached to the airframe with cement or epoxy if not otherwise supported. This will prevent noise problems or erratic operation that could occur if the plane moves. Capacitance will occur as an antenna base is separated from its ground plane. This may distort antenna coverage or operation.

FIGURE 3-2. ANTENNA GROUND PLANE FOR NONCONDUCTIVE MOUNT

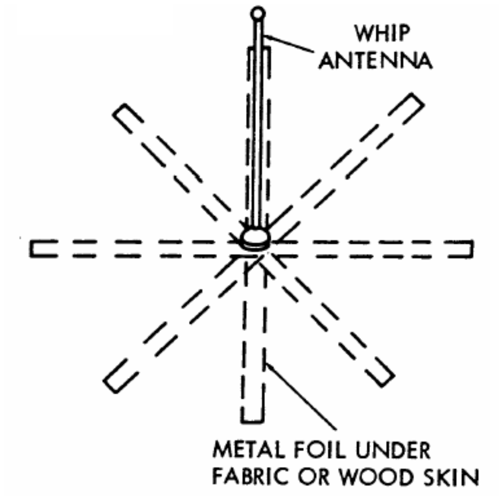


FIGURE 3-3. ONE MEANS TO PROVIDE ADEQUATE ANTENNA BONDING THROUGH A COMPOSITE AIRFRAME

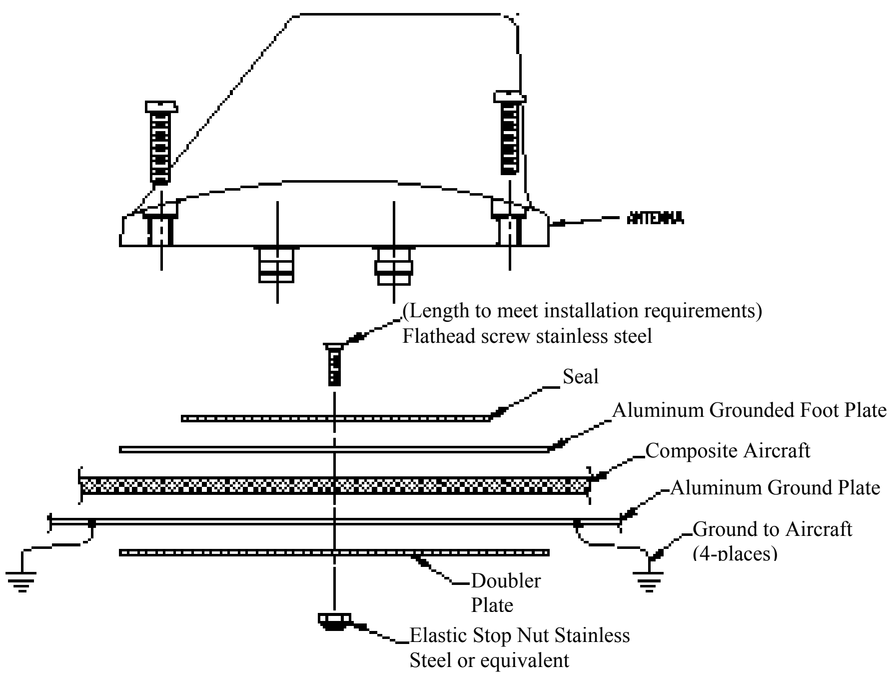
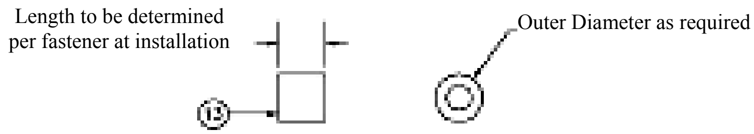


FIGURE 3-4. USE ROLLED ALUMINUM AS BUSHINGS FOR ALL SCREWS



NOTE: Carbon Fiber composite material, while conductive has not been found to be adequate as a ground plane.

311. ANTENNA FEED LINE BALUN.

a. Antenna cables (electrical feed lines) may be designed to be electrically balanced or unbalanced. Certain antenna designs incorporate dual elements with a requirement for balanced input. In these cases, standard cables which are unbalanced are generally employed for the cable run and a balancing transformer is located at the antenna feed connection.

b. A balun is a device that converts an unbalanced feed into a balanced input and may include a transformer that matches feed impedance to provide maximum signal transfer. Follow the manufacturer's installation procedures when a balun is required. Some balun designs require that the balun be grounded to the airframe.

c. Refer to AC 43.13-1, chapter 11 (as amended), for bonding practices.

FIGURE 3-5. TYPICAL DIPOLE ANTENNA ASSEMBLY

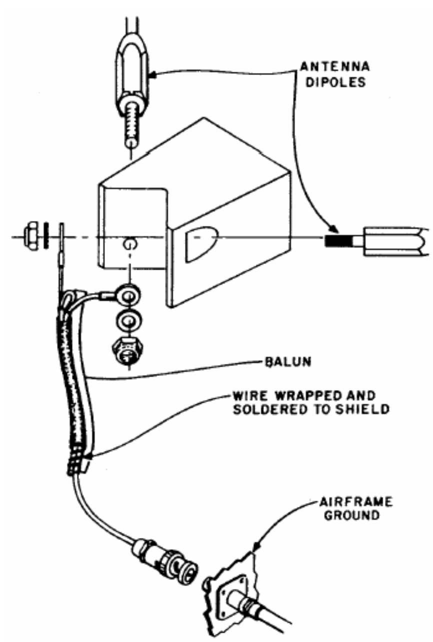
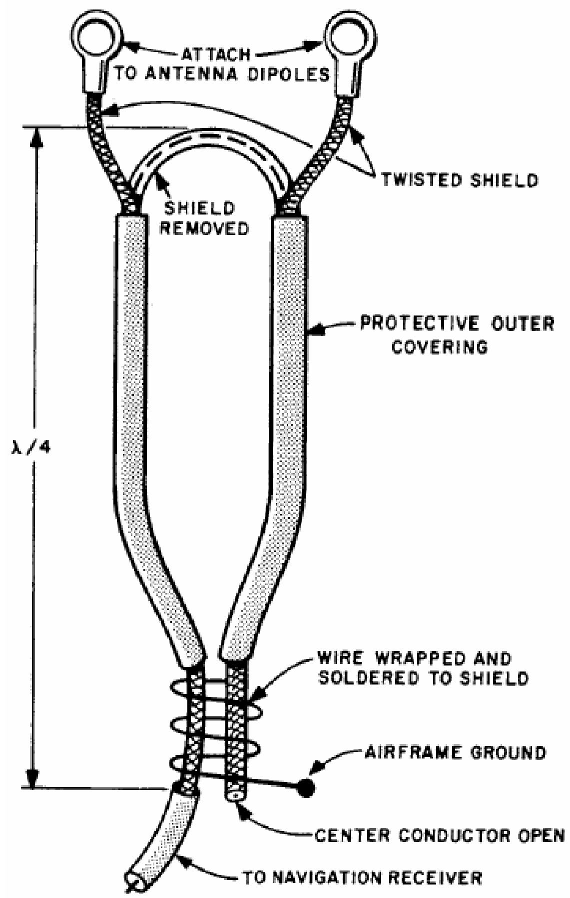


FIGURE 3-6. TYPICAL VOR BALUN



312. ANTENNA REPAIR. Painting an antenna or applying protective coatings or devices that are not approved are not allowed under this AC. Paint is a RF de-tuner. If an antenna is painted in the field, paint type and paint thickness present uncontrolled variables that will affect an antenna's performance, and may result in the antenna no longer meeting its specifications or Technical Standard Order.

313. THRU 399. RESERVED

FIGURE 3-7. ADF LOOP

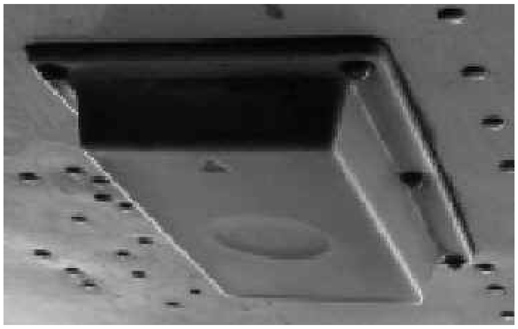


FIGURE 3-8. ADF COMBINED SENSE LOOP

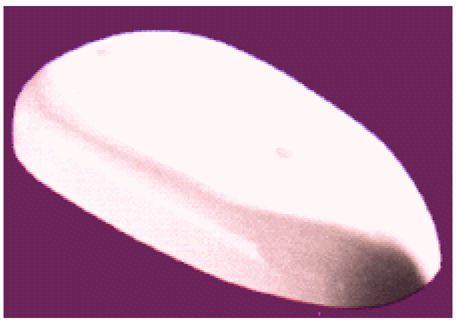


FIGURE 3-9. COM WHIP

FIGURE 3-10. COM WHIP BENT

FIGURE 3-11. ELT

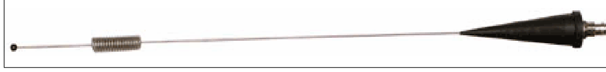


FIGURE 3-12. GPS

FIGURE 3-13. GLIDESLOPE

FIGURE 3-14. MARKER

FIGURE 3-15. COMBINED COM/VOR

FIGURE 3-16. VOR RABBIT EAR

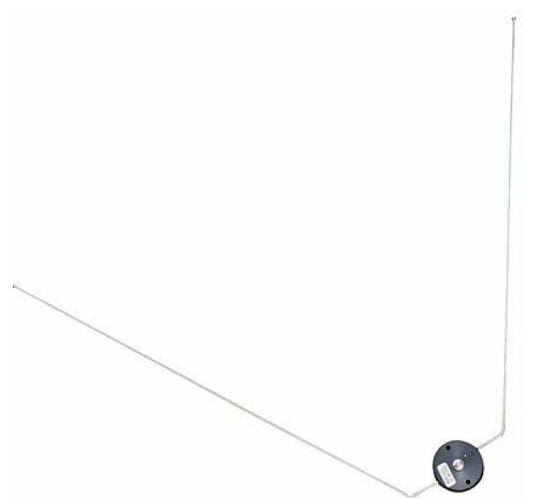


FIGURE 3-17. VOR BLADES

FIGURE 3-18. TRANSPONDER/DME PROBE

FIGURE 3-19. DME/TRANSPONDER BLADE

CHAPTER 4. ANTICOLLISION AND SUPPLEMENTARY LIGHT INSTALLATION

400. ANTICOLLISION AND SUPPLEMENTARY LIGHT SYSTEMS.

a. Anticollision Lights. The requirements for anticollision lights are included in Title 14 of the Code of Federal Regulations (14 CFR) part 23, § 23.1401 and part 27, § 27.1401 for non-transport category aircraft. Aircraft for which an application for type certificate was made before April 1, 1957, may conform either to the above regulations or to the following standards:

(1) Anticollision lights (when installed) should be installed on top of the fuselage or tail in such a location that the light will not impair the flight crewmembers' vision and will not detract from the conspicuity of the position lights. If there is no acceptable location on top of the fuselage or tail, a bottom fuselage or wing tip installation may be used.

(2) The color of the anticollision light shall be either aviation red or aviation white in accordance with the specifications of § 23.1397 or § 27.1397 as applicable.

(3) The arrangement of the anticollision light system, (i.e., number of light sources, beam width, speed of rotation and other characteristics, etc.) must give an effective flash frequency of not less than 40, nor more than 100, cycles per minute. The effective flash frequency is the frequency at which the aircraft's complete anticollision light system is observed from a distance, and applies to each sector of light including any overlaps that exist when the system consists of more than one light source. In overlaps, flash frequencies may secede 100, but not 180 cycles per minute.

(4) The system must consist of enough lights to illuminate the vital areas around the aircraft, considering the physical configuration and flight characteristics of the aircraft. The field of coverage must extend in each direction within at least 75 degrees above and 75 degrees below the horizontal plane of the aircraft, except that there may be solid angles of obstructed visibility totaling not more than 0.5 steradians.

(5) The minimum light intensity and minimum effective intensities are given in §§ 23.1401 and 27.1401 respectively. These intensities are given in a table and formula format for both part 23 and § 27 aircraft.

b. Supplementary lights may be installed in addition to position and anticollision lights required by applicable regulations; provided that, the required position and anticollision lights are continuously visible and unmistakably recognizable and their conspicuity is not degraded by such supplementary lights.

401. INTERFERENCE.

a. Crew Vision. Partial masking of the light may be necessary to prevent direct or reflected light rays from any anticollision or supplementary light from interfering with crew vision. Determine if the field of coverage requirements are met. An acceptable method of preventing light reflection from propeller disc, nacelle, or wing surface is an application of nonreflective paint on surfaces which present a reflection problem. Perform a night flight-check to assure that any objectionable light reflection has been eliminated. Enter a notation to that effect in the aircraft records.

b. Communication and Navigation. Assure that the installation and operation of any anticollision/supplementary light does not interfere with the performance of installed communication or navigation equipment. Capacitor discharge light (strobe) systems may generate radio frequency interference (RFI). This radiated interference can be induced into the audio circuits of communication or navigation systems and is noticeable by audible clicks in the speaker or headphones. The magnitude of the RFI disturbance does not usually disrupt the intelligence of audio reception.

c. Precautions. RFI can be reduced or eliminated by observing the following precautions during installation of capacitor discharge light systems:

(1) Locate the power supply at least 3 feet from any antenna, especially antennas for radio systems that operate in the lower frequency bands.

(2) Assure that the lamp unit (flash tube) wires are separated from other aircraft wiring placing particular emphasis on coaxial cables and radio equipment input power wires.

(3) Make sure that the power supply case is adequately bonded to the airframe.

(4) Ground the shield around the interconnecting wires between the lamp unit and power supply at the power supply end only.

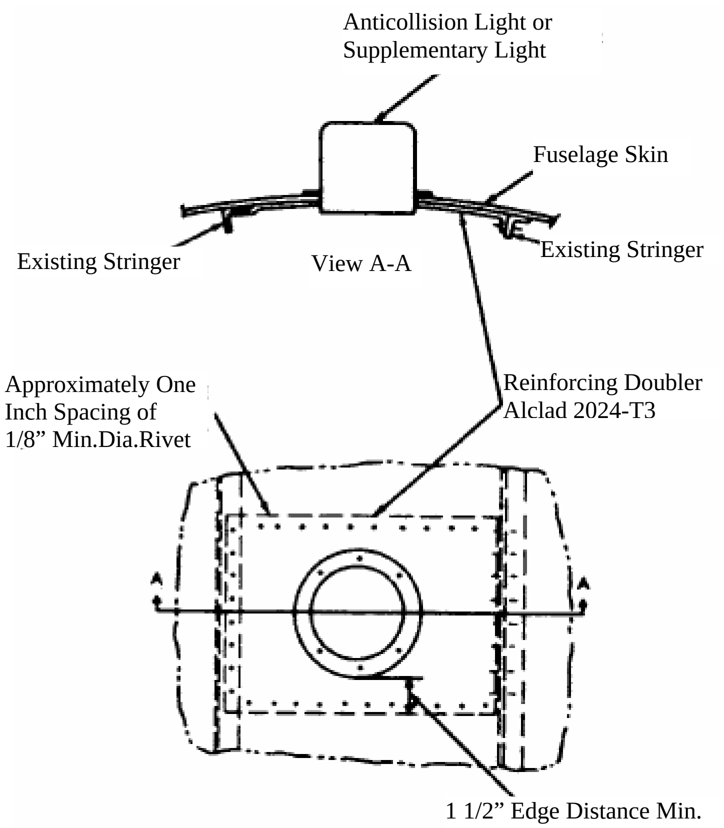
402. MARKINGS AND PLACARDS. Identify each switch for an anticollision/supplementary light and indicate its operation. The aircraft should be flight tested under haze, overcast, and visible moisture conditions to ascertain that no interference to pilot vision is produced by operation of these lights. If found, unsatisfactory by test or in the absence of such testing, a placard should be provided to the pilot stating that the appropriate lights be turned off while operating in these conditions.

403. ELECTRICAL INSTALLATION. Install an individual switch for the anticollision light or supplementary light system that is independent of the position light system switch. Data for the installation of wiring, protection device, and generator/alternator limitations is contained in Advisory Circular (AC) 43.13-1, Acceptable Methods, Techniques, and Practices-Aircraft Inspection and Repair, (as amended) chapter 11. Assure that the terminal voltage at each light is within the limits as prescribed by the manufacturer.

404. ALTERATION OF STRUCTURE.

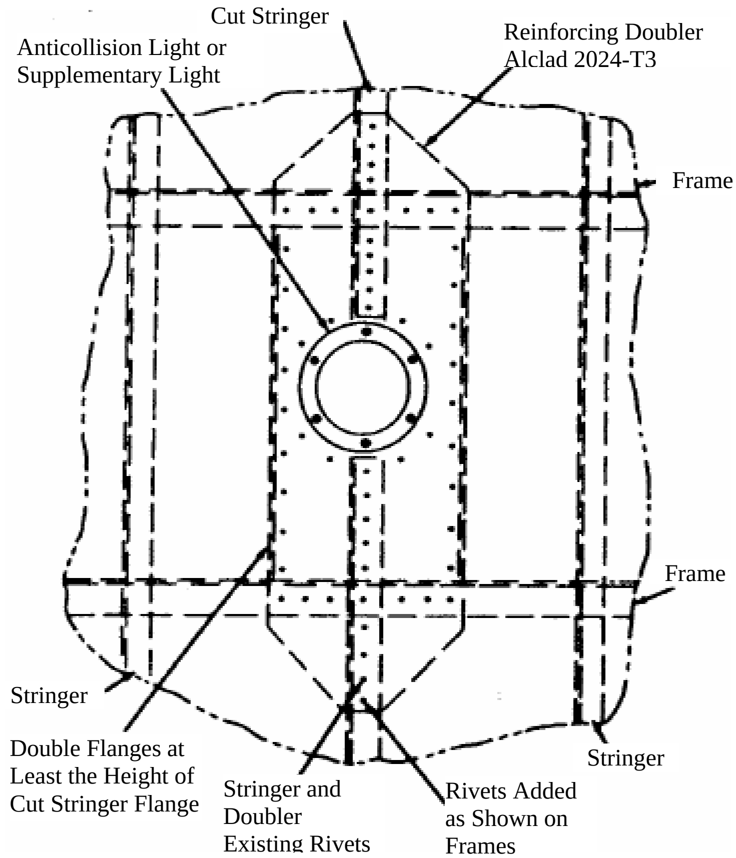
a. The simplest light installation is to secure the light to a reinforced fuselage skin panel. The reinforcement doubler shall be of equivalent thickness, material, and strength as the existing skin. (Install as shown in Figure 4-1.)

FIGURE 4-1. TYPICAL ANTICOLLISION OR SUPPLEMENTARY LIGHT INSTALLATION IN A SKIN PANEL (UNPRESSURIZED)



b. When a formed angle stringer is cut and partially removed, position the reinforcement doubler between the skin and the frame. The doubler is to be equivalent to the stringer in thickness and extend lengthwise beyond the adjacent fuselage frames. The distance between the light and the edge of the doubler is to be twice the height of the doubler flange. (See Figure 4-2 for typical installation.)

FIGURE 4-2. TYPICAL ANTICOLLISION OR SUPPLEMENTARY LIGHT INSTALLATION INVOLVING A CUT STRINGER (UNPRESSURIZED)

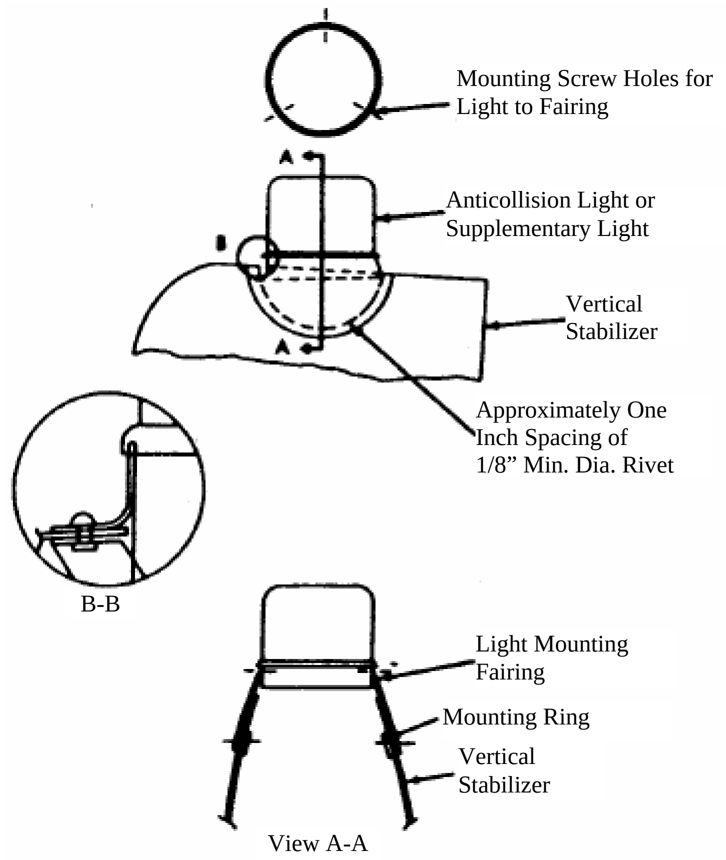


c. Engineering evaluation is required for installations involving the cutting of complex formed or extruded stiffeners, fuselage frames, or pressurized skin of pressurized aircraft.

d. Vertical stabilizer installations may be made on aircraft if the stabilizer is large enough in cross section to accommodate the light installation, and if aircraft flutter and vibration characteristics are not adversely affected. Locate such an installation near a spar, and add formers as required to stiffen the structure near the light. (A typical installation is shown in Figure 4-3.)

e. Rudder installations are not recommended because of the possible structural difficulties. However, if such installations are considered, make an engineering evaluation to determine whether the added mass of the light installation will adversely affect the flutter and vibration characteristics of the tail surfaces.

FIGURE 4-3. TYPICAL ANTICOLLISION OR SUPPLEMENTARY LIGHT INSTALLATION IN A FIN TIP



NOTE: Skin thickness of mounting ring and fairing are at least equivalent.

405. GUIDELINES FOR INSTALLATIONS.

a. Prior Approval. Due to the complexity of measurements for intensity, field of coverage, and color, evidence of meeting §§ 23.1401, 27.1401, or FAA approval should be obtained from the light manufacturer before installation.

b. System Performance.

(1) Field of coverage. Evidence of meeting §§ 23.1401, 27.1401, or FAA approval for "field of coverage" should be obtained from the light manufacturer before installation. To insure that the manufacturer’s approved field of coverage is applicable to an installation, this mounting tolerance should not be exceeded.

(2) Obstructed visibility. Measure all solid angles of obstruction within the required field of coverage. For multiple light installations, coverage between the mounting levels is not necessary. When a multiple light installation is being evaluated, shadows for each light should be measured independently, and only shadow areas repeated in each independent measurement (overlap) should be counted. Methods for determining the amount of obstructed visibility are given below; however, other methods can give acceptable results.

(a) Wall shadows. This procedure is applicable to installations where shadows from light obstructions appear on a vertical surface such as a hangar, wall. Validity is based on two facts: (1) that a vertical surface can approximate a sphere surface if the distance from the light is considerable, and the shadow is reasonably small, and (2) that sphere surface area can be converted to steradians by dividing by the radius squared.

1. Position the aircraft in a darkened hangar so that the longitudinal axis is perpendicular to a hangar wall. Level for weight and balance. To keep measure errors low, the distance from light to wall should be as great as practicable considering hangar size. The distance should not be less than 20 feet.

2. Turn on the lights and measure the area of wall shadows. Sufficient points should be marked and identified so that the shadow pattern can be transferred to graph paper for accurate evaluation. The area can be found by counting squares on the graph or by using a planimeter. Measurements should include areas of transition from shadowed to lighted areas. For top light measurements on multiple light installations, only shadows above the level of the top light should be considered.

3. Compute the solid angle obstruction in steradians, by dividing each shadow area by the square of the distance from the center of the area to the light.

4. Evaluate the results to determine if the system consists of enough lights to illuminate the vital areas around the aircraft, considering the physical configuration and flight characteristics of the aircraft. The field of coverage must extend in each direction within at least 30° above and 30° below the horizontal plane of the aircraft, except that there may be solid angles of obstructed visibility totaling not more than the requirements of § 23.1401 or § 27.1401.

5. For installations where shadows are restricted to directly aft and centered about the longitudinal axis, the following procedures apply:

A. Establish a point on the wall, which corresponds to a line parallel to the longitudinal axis and through the light associated with the shadow.

B. Measure the distance from the light to the point and determine the area representing 0.15 steradians (A=0.15d2). The distance (d) should be at least 20 feet.

(b) Ramp shadows. This procedure is applicable to shadows which appear on a horizontal plane such as a flat level ramp and will be associated with a top mounted light and a 0.5 steradian limit. Area measurements as described in the wall shadow method should not be used. Some error is inherent, because horizontal angles are measured on a plane displaced from the light source. To compensate for these measurement errors, Table 4-1 is furnished to convert from measured solid angles to true solid angles. A term "square degrees" is used to aid in the discussions of solid angle measurements.

1. If no masking is required, remove the red cover and attach its clamp ring to the light base. If masking is used, obtain a clear cover and install it with a duplicate mask.

2. Center the aircraft on the largest available dark ramp. A minimum of 50 feet radius of clear ramp space will usually be needed. If the installation is symmetrical, clear ramp space will be needed on one side only. Level the aircraft as for weight and balance check. Trim the flaps, rudder, elevator, and ailerons. With jacks in place, raise the gear if the measurement results would otherwise be affected.

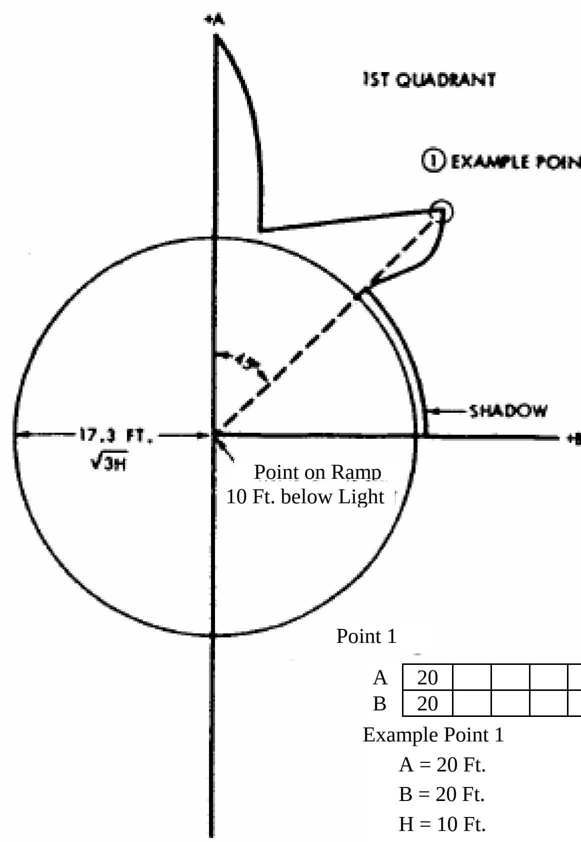
3. Chalk the following marks on the ramp:

A. A reference point directly below the light.

B. A circle centered on this reference point having a radius equal to 1.732 times the height of the light above the point. (The circle represents area beyond the minus 30° vertical limit and does not require lighting.)

C. A line parallel to the aircraft longitudinal centerline, which passes through the reference point.

FIGURE 4-4. RAMP SHADOW PATTERN



Point 1

A	20									
B	20									

Example Point 1

A = 20 Ft.
B = 20 Ft.
H = 10 Ft.
Distance to point 1 = 28.2 Ft.
Horizontal Angle = 45˚
Vertical Angle = 19.4˚

D. A line perpendicular to the above line, which also passes through the reference point.

4. At night, turn on the anticollision light and chalk all shadow patterns (except jack shadows), which appear outside the circle. If the light rotates so as to cause the shadows to oscillate, lay twine along the outermost edge of the shadow, and chalk along the line.

5. Move the aircraft to facilitate area measurements. Measure, sketch, and chart the ramp marks and other information as shown in Figure 4-4. Measure enough points along the shadow patterns to accurately describe them. Make enough sketches to include all shadows.

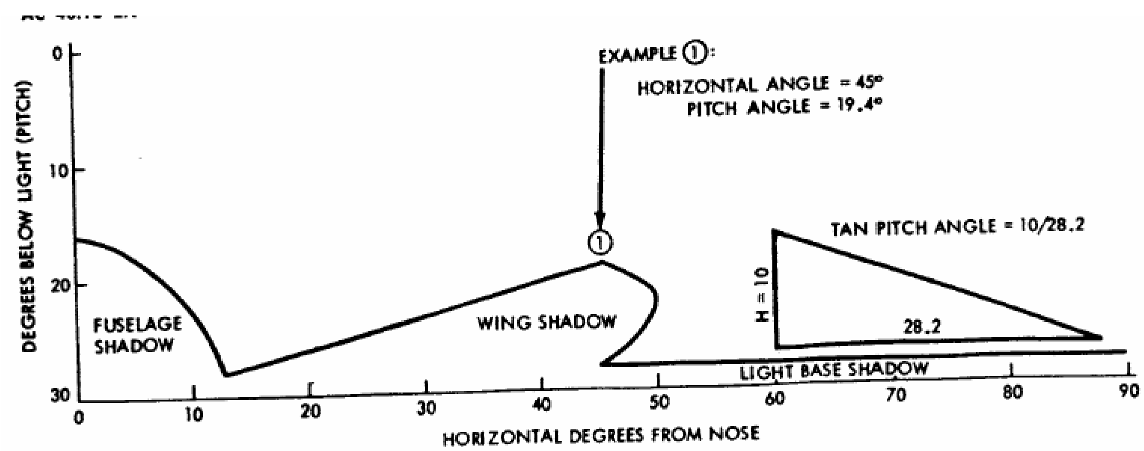
6. Convert the above measurements to a graph showing vertical degrees vs. horizontal degrees as shown in Table 4-1 (first quadrant). The second quadrant will have to be measured also to obtain the total shadow area for one side.

7. If the shadow pattern is symmetrical, no other measurements will be necessary. Use 1 degree for each space on the graph paper and count the square degrees of shadow. A total of 1,642 or less is within limits. A total of 1,872 or more is out of limits, and, if possible, should be reduced by adding or moving a light, or by trimming a mask. If the count is between 1,642 and 1,872, proceed as follows: For each 1 degree segment of vertical (pitch), convert the counted square degrees to true square degrees using Table 4-1. If the sum of the true square degrees from all segments exceeds 1,642 (0.5 steradian), the installation is out of limits.

(c) Scale drawings. Accurate scale drawings can be used to measure solid angles of obstruction. Such drawings should have sufficient size and accuracy to give dependable results. In some cases, actual measurements can be combined with small drawings as shown in Figure 4-6. For the 6 points established on the left wing, a string can be used to connect the light successively to each. A protractor can then be used to measure the vertical angle from level. The horizontal angle for each point can be measured on the top view (center). When both horizontal and vertical angles for each point have been determined, they can be plotted on a graph as shown in Figure 4-7. If a symmetrical condition exists, only the first and second quadrants need be measured.

1. The first quadrant contains approximately 450 square degrees of obstruction. The other wing quadrant will double this to 900 square degrees.

FIGURE 4-5. CONVERSION TO VERTICAL VS. HORIZONTAL DEGREES



2. A measurement of the fin and rudder shadows adds approximately 100 for a total of 1,000 square degrees. Since a maximum of 1,642 square degrees is allowed (0.5 steradians), this installation is well within limits. Due to limitations of the method, results within 10 percent (0.05 steradians) of the limit are questionable. Many times, a mask is required to prevent reflections into the cockpit. In. the example of Figures 4.9 and 4.10, the installed mask blocks the light for ±10 horizontal degrees and from -10 to -30 vertical degrees. These 400 square degrees were measured at the light.

3. When larger drawings are used and no actual aircraft measures are made, vertical angles should not be taken directly from the drawing, but should be computed as follows:

A. On the aircraft side view, measure the vertical distance from a point to the light level.

B. On the top view, measure the distance from the point to the light.

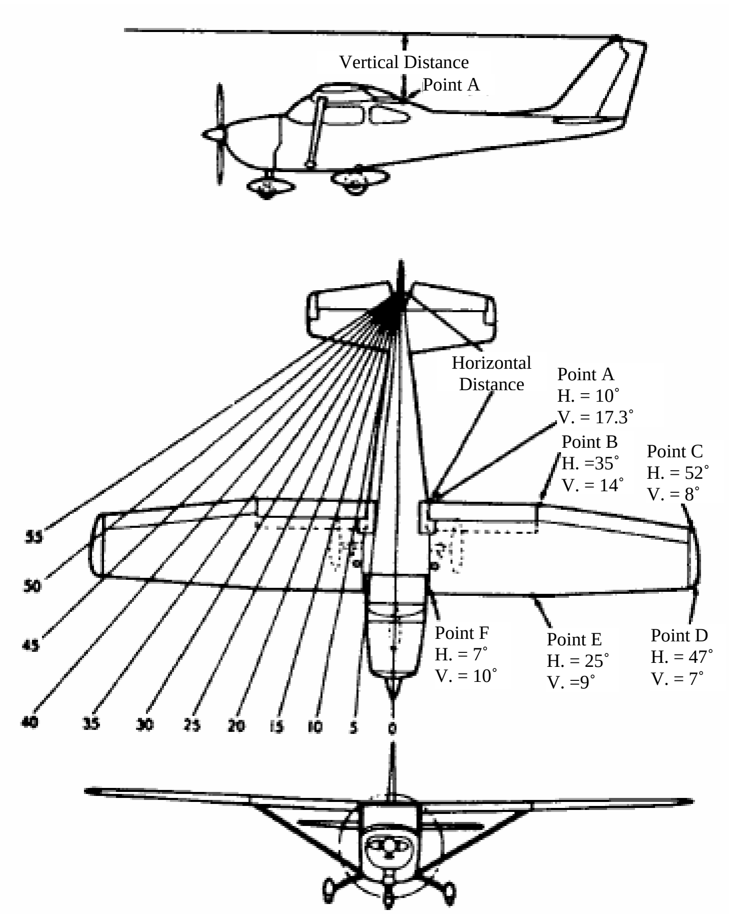
C. Compute:

Tangent of vertical angle=vertical distance/ horizontal distance

TABLE 4-1. CONVERSION TO TRUE SQUARE DEGREES—FIRST QUADRANT

Pitch Segment (Degrees)	Measured Square Degrees	Correction Factor	True Square Degrees	Pitch Segment (Degrees)	Measured Square Degrees	Correction Factor	True Square Degrees
30-29	90	.87036	78.33	15-14		.96815	
29-28	90	.87882	79.09	14-13		.97237	
28-27	44	.88701	39.03	13-12		.97630	
27-26		.89493		12-11		.97992	
26-25		.90259		11-10		.98325	
25-24		.90996		10-9		.98629	
24-23		.91706		9-8		.98902	
23-22		.92388		8-7		.99144	
22-21		.93042		7-6		.99357	
21-20		.93667		6-5		.99540	
20-19		.94264		5-4		.99692	
19-18		.94832		4-3		.99813	
18-17		.95372		3-2		.99905	
17-16		.95882		2-1		.99966	
16-15		.96363		1-0		.99996	
						TOTAL	

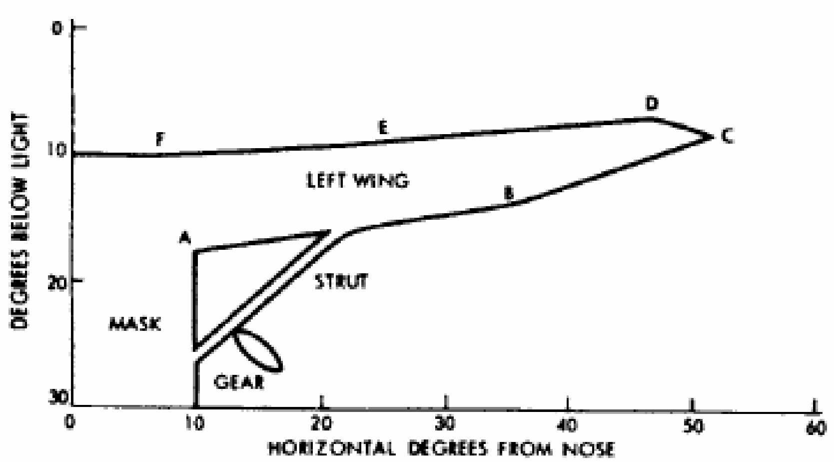
FIGURE 4-6. SCALE DRAWINGS



(d) Flashing characteristic.

1. Turn off any flashing supplementary lighting. Observe the flashing of the anticollision light system at a point where each light can be observed independently, and determine that each flashing rate is between 40 and 100 flashes per minute.

FIGURE 4-7. SOLID ANGLE BLOCKAGE—FIRST QUADRANT



2. For multiple light systems, observe at a point where overlap occurs, and determine that the combined flashing rate does not exceed 180 flashes per minute. Flashing outside the required field of coverage is not necessary.

406. THRU 409. RESERVED.

CHAPTER 5. SKI INSTALLATIONS

500. PURPOSE. This chapter provides information for ski installations on small aircraft.

501. HAZARDS AND WARNINGS. Operation of ski planes exposes the aircraft and its occupants to additional risks not associated with landplanes. The additional weight and size of skis place additional ground and air loads on the aircraft. Ground handling, taxi, takeoff, and landing can place significant side loads and twisting moments on landing gear and attachment structure which can cause hidden or cumulative damage. Improper rigging, and weak springs or shock cords can allow the front of skis to "dump" or rotate downward, in flight, possibly rendering the aircraft uncontrollable. Skis with weak springs or shock cords may rotate tip-down and "dig" or penetrate the snow, during takeoff or landing on rough or drifted snow, which could result in an accident. In-service failure of attachment hardware, springs, shock cords, or cables poses great risk of entering the propeller arc, which has resulted in complete loss of aircraft in flight. For these reasons, proper installation, rigging, periodic inspection, and maintenance are of utmost importance to safety. Consultation with experienced ski maintenance technicians and operators is strongly recommended when considering any ski installation or alteration.

502. ADDITIONAL REFERENCE MATERIAL.

a. Airframe and Ski manufacturers' data, if available.

b. Advisory Circular (AC) 43.13-1, Acceptable Methods, Techniques, and Practices – Aircraft Inspection and Repair.

c. AC 43-210, Standardized Procedures for Requesting Field Approval of Data, Major Alterations, and Repairs.

d. FAA Order 8110.54, Instructions for Continued Airworthiness.

e. FAA-H-8083-23, Seaplane, Ski plane, and Float/Ski Equipped Helicopter Operations Handbook.

503. INSTALLATION CONSIDERATIONS.

a. Determining Eligibility of Aircraft. Only aircraft approved for operation on skis are eligible for ski installations in accordance with this chapter. Eligibility can be determined by referring to the Aircraft Specifications, Type Certificate Data Sheets (TCDS), Aircraft Listing, Summary of Supplemental Type Certificates, or by contacting the manufacturer. Also determine the need for any required alterations to the aircraft to make it eligible for ski operations. (See ski plane-specific entries throughout TCDS A4CE on airspeeds, weight and balance limitations and additional placards, and the Required Equipment listing for the first model in Aircraft Specification A-790 for examples.) If the aircraft is not approved in a ski plane configuration by type design, then skis cannot be installed by relying solely on data in this AC. Contact FAA engineering for approval or obtain/develop approved data from another source.

b. Identification of Approved Model Skis. Determining that the skis are an approved model can be done by referring to the identification plate or placard displayed on the skis. A Technical Standard Order (TSO) number; Type Certificate (TC) number; or an aircraft part number, if the skis have been approved as a part of the aircraft, will be shown thereon if the skis are approved models.

c. Maximum Limit Load Rating.

(1) Approved ski installations. Before installation, determine that the maximum limit load rating (L) as specified on the ski identification plate or placard is at least equal to the maximum static load (S) times the limit landing load factor (η) previously determined from static drop tests of the airplane by the aircraft manufacturer, represented by the following equation:

$$L = S \times \eta$$

(2) Unknown limit landing load factor. In lieu of a value η determined from such drop tests, a value of η determined from the following formula may be used:

$$\eta = 2.80 + \frac{9000}{W + 4000}$$

where "W" is the certificated gross weight of the airplane

d. Oversize Ski Installations. Skis approved for airplanes of greater gross weight than the airplane on which they are to be installed may be used provided the geometry of the ski is similar to that of a ski previously approved for the airplane (not more than 10 percent difference in width or length of contact surface). This limitation is to assure that the performance of the airplane will not be adversely affected by oversize skis.

e. Landing Gear Moment Reactions. In order to avoid excessive moment reactions on the landing gear and attachment structure, the ski pedestal height must not exceed 130 percent of the axle centerline height with the wheel and tire installed.

504. FABRICATION AND INSTALLATION.

a. Hub-Axle Clearance. The pedestal hub should fit the axle to provide a clearance of .005" minimum to .020" maximum. Hubs may be bushed to adjust for axle size, using any ferrous or nonferrous metal, hard rubber, or fiber. If rubber or fiber bushings are used, use retaining washers of sufficient size on each side to retain the hub if the bushing should slip or fail. (See Figure 5-1.) Field experience has shown that use of good quality, low-temperature grease, particularly modern synthetic-based grease, improves ski operation and wear protection when used on the axle-to-hub or axle-to-bushing faying surfaces.

FIGURE 5-1. TYPICAL HUB INSTALLATION

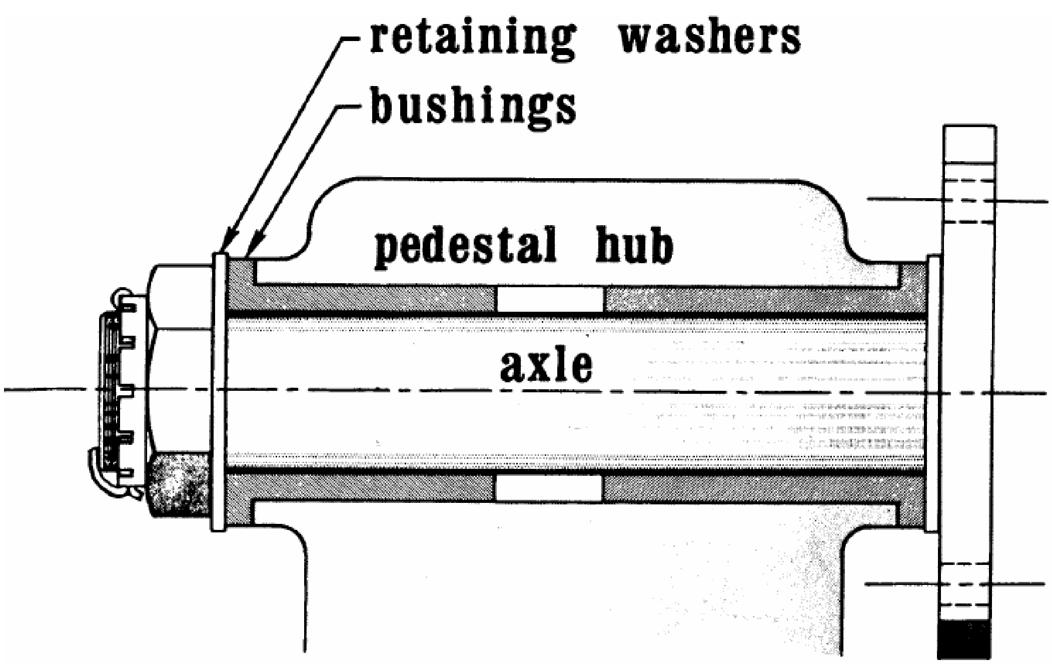
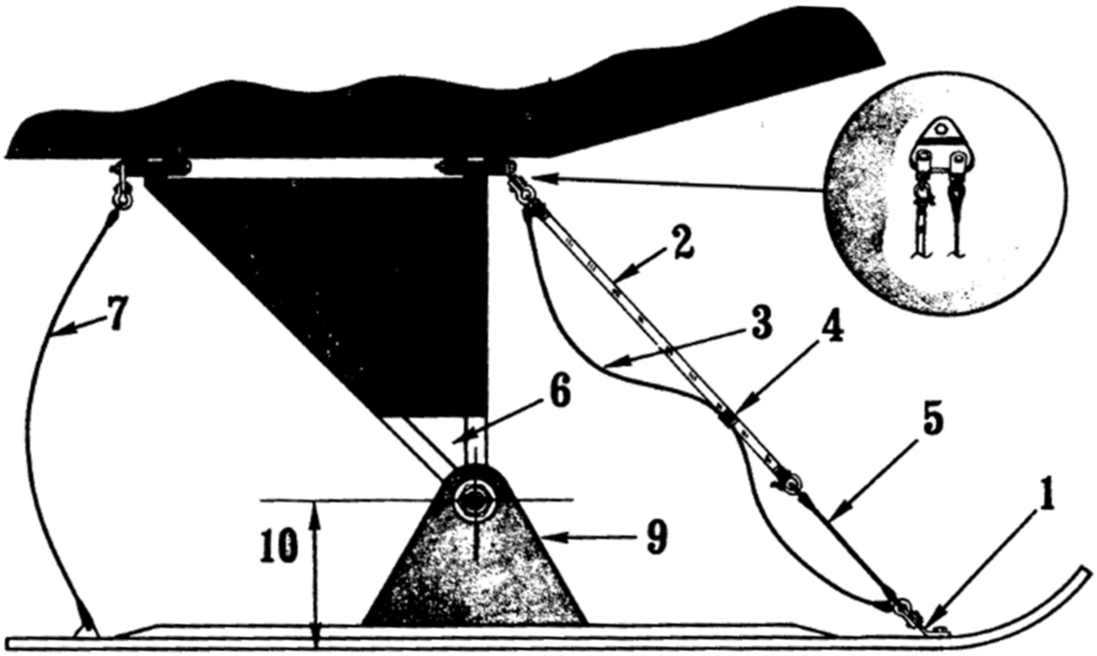


FIGURE 5-2. TYPICAL SKI INSTALLATION



1. Fitting
2. Shock Cord
3. Safety Cable
4. Tape
5. Crust-cutter Cable
6. Fabric removed to facilitate inspection
7. Check Cable
8. Clevis
9. Ski Pedestal
10. Pedestal Height

b. Crust-Cutter Cables. Crust-cutter cables are optional. However, when operating in severe crust conditions, it is advisable to have this cable installed to prevent the shock cord from being cut if the nose of the ski breaks through the crust while taxiing.

c. Cable and Shock Cord Attachment and Attachment Fittings.

(1) Field experience. Service reports indicate that failure of the ski itself is not a predominant factor in ski failures. Rigging (improper tension and terminal attachments) and cast-type pedestal material failures are predominant. Failures of the safety cable and shock cord attachment fittings usually occur at the ski end and not at the fuselage end.

(2) Separating attachment points. Do not attach tension cords and safety cables at the same point on the fuselage fittings. Provide separate means of attaching cables and shock cords at the forward and aft ends of the skis.

(3) Fabrication. Approved skis are normally supplied with cables, shock cord, and fittings; however, the specifications in Table 5-1 may be used for their fabrication and installation.

NOTE: Field experience indicates that accelerated wear and damage can occur to a 1/8" cable and its attachment hardware in normal service on skis with at or above a 1500-pound Limit Load Rating. We recommend a minimum of 5/32" cable be used at or above this rating. A 1/8" cable may be suitable for use on aircraft with light-weight skis and less that 1500-pound maximum certificated gross weights, such as those meeting the definition of Light-Sport Aircraft.

TABLE 5-1. MINIMUM CABLE AND SHOCK CORD SIZES

Ski Limit Load Rating	Single Safety Cable	Double Safety Cable	Single Crust-Cutting Cable	Double Crust-Cutting Cable	Single Shock Cord	Double Shock Cord
Up to-3000	1/8”	1/8”	1/8”	1/2”	1/2”	1/2”
3000-5000	-----	1/8”	-----	1/8”	-----	1/2”
5000-7000	-----	5/32”	5/32”	5/32”	3/4”	3/4”
7000-9000	-----	3/16”	-----	5/32”	-----	3/4”

(a) Make check cable, safety cable, and crust-cutting cable ends by the splice, swage, or nicopress methods. Cable clamps may be used if adjustable lengths are desired, but they are not recommended. Use standard aircraft hardware only. (Hardware used to attach cables must be compatible with cable size.) Refer to AC 43.13-1, chapter 7, as amended, for more information on cable fabrication.

(b) Shock cord ends may be fabricated by any of the following methods:

1. Make a wrapped splice using a proper size rope thimble and No. 9 cotton cord of .035” (minimum) safety wire. Attach with clevis or spring steel snap fastener. (*Do not* use cast iron snaps.)

2. Use approved spring-type shock cord end fasteners.

(c) Fitting (Figure 5-3) Specifications and Installation:

1. Fittings fabricated for 1/8-inch cable or 1/2-inch shock cord shall be at least .065” 1025 steel or its equivalent.

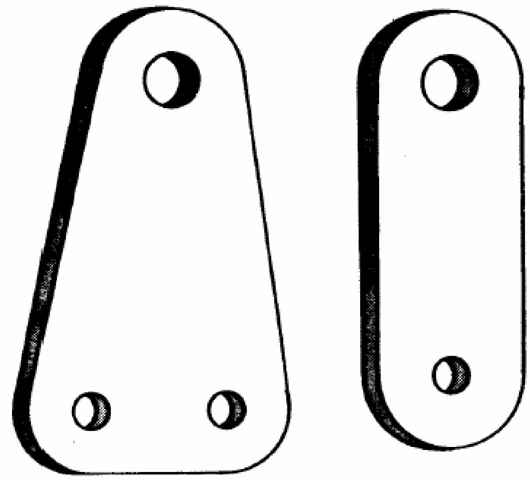
2. Fittings fabricated for 5/32-inch cable or 3/4-inch shock cord shall be at least .080” 1025 steel or its equivalent.

3. An improperly installed fitting may impose excessive eccentric loads on the fitting and attach bolts and result in failure of the fitting or bolts.

4. If attaching cables directly to holes in fittings, radius the hole edges to reduce stress concentration and accelerated wear of the thimble. Stainless steel thimbles are recommended for increased wear resistance.

5. If attaching cables to fittings using clevises, clevis bolt castellated nuts should be used, then properly torqued and safetied with cotter pins. Field experience has shown that diaper-pin-style quick-releasing safety devices are more prone to failure during operation, and are not recommended.

FIGURE 5-3. TYPICAL FUSELAGE FITTINGS



d. Provisions for Inspection. Aircraft using fabric-covered landing gear should have at least the lower 4 inches of fabric removed to facilitate inspection of the axle attachment area, and to prevent the entrapment of snow and ice, which can lead to damage and corrosion of the landing gear. (See Item 6 in Figure 5-2.)

505. RIGGING OF SKIS.

a. Location of Attach Fittings on Fuselage or Landing Gear. Locate fittings so the shock cord and cable angles are not less than 20 degrees when measured in the vertical plane with the shock absorber in the fully extended position (see Angle B, Figures 5-4 and 5-5).

NOTE: Do not attach fittings to wing-brace struts, except by special approval (manufacturer or FAA).

b. Main Ski Incidence Angles.

(1) Set cable lengths with the aircraft level and shock absorbers fully extended.

(2) Adjust length of check cable to provide a +0- to +5-degree ski incidence angle (reference Figures 5-4 and 5-6).

(3) Adjust length of safety cable to provide a -15-degree ski incidence angle (reference Figures 5-5 and 5-6).

FIGURE 5-4. MAIN SKI AT MAXIMUM POSITIVE INCIDENCE (CHECK CABLE TIGHT)

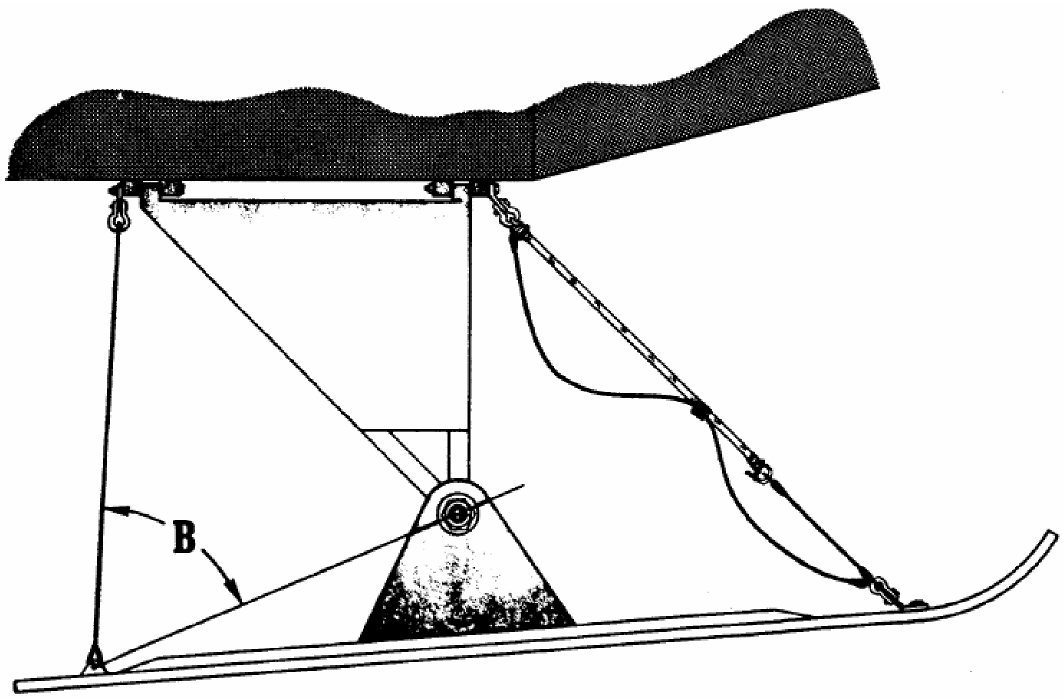


FIGURE 5-5. MAIN SKI AT MAXIMUM NEGATIVE INCIDENCE (SAFETY CABLE TIGHT)

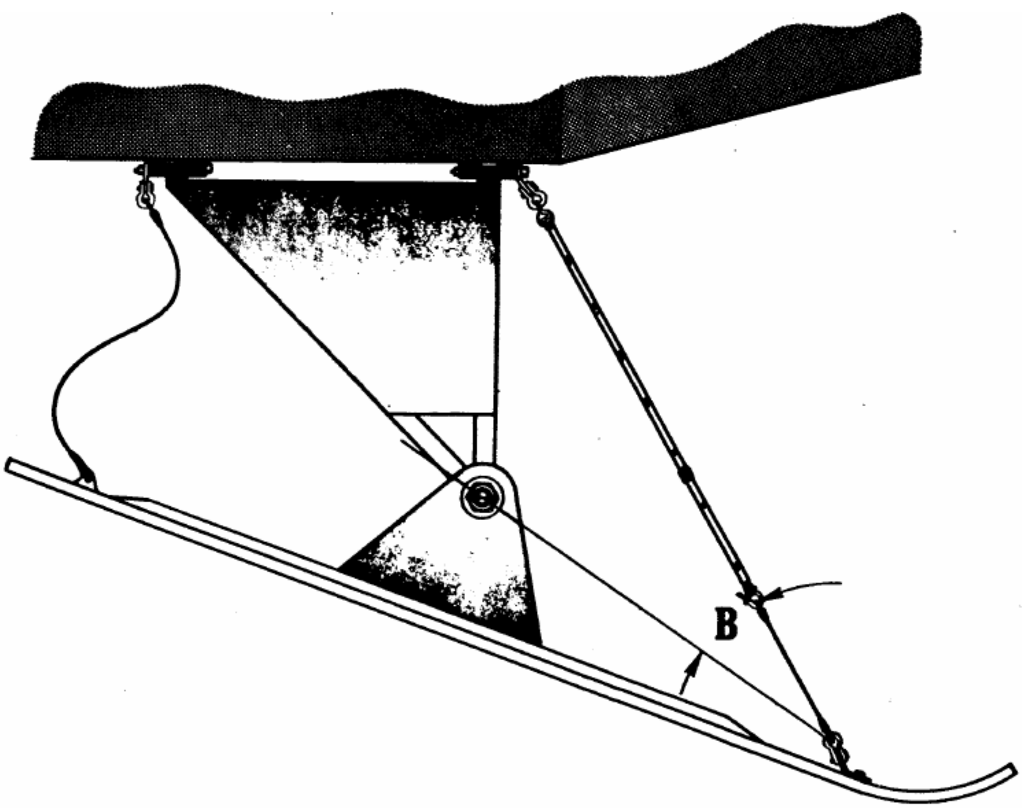
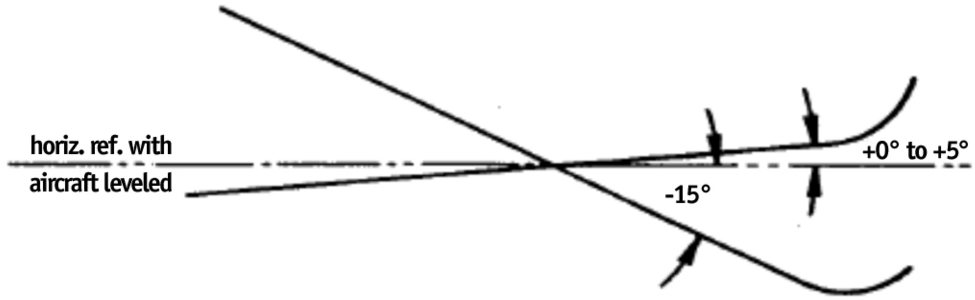


FIGURE 5-6. MAIN SKI INCIDENCE ANGLES



c. Tension Required in Main Ski Shock Cords. Apply sufficient shock cord tension to fore end of the skis to prevent flutter at various airspeeds and attitudes. Because of the various angles used in attaching the shock cord to the skis, shock cord tension cannot be specified. In most installations on rigid, truss type landing gear, the downward force applied at the fore end of the ski, sufficient to cause the check cable to slacken, should be approximate to the listing in Table 5-2.

TABLE 5-2. APPROXIMATE MAIN SKI TENSION FORCES

Ski Limit Load Capacity	*Downward Force (pounds)*
1500-3000	*20-40*
3000-5000	*40-60*
5000-7000	*60-120*
7000-9000	*120-200*

NOTE: Do not rely upon these downward force values on aircraft with spring steel or other flexible landing gear. Due to variations in gear leg flexibility, a generic tension force table is difficult to establish.

d. Springs in place of Shock Cords. If springs are used in place of shock cords to provide rigging tension, they must be able to withstand extreme cold and slight external scratching without premature fatigue failure, and must not cause skis, rigging, or landing gear to experience flutter or objectionable vibration during aircraft flight and dive tests.

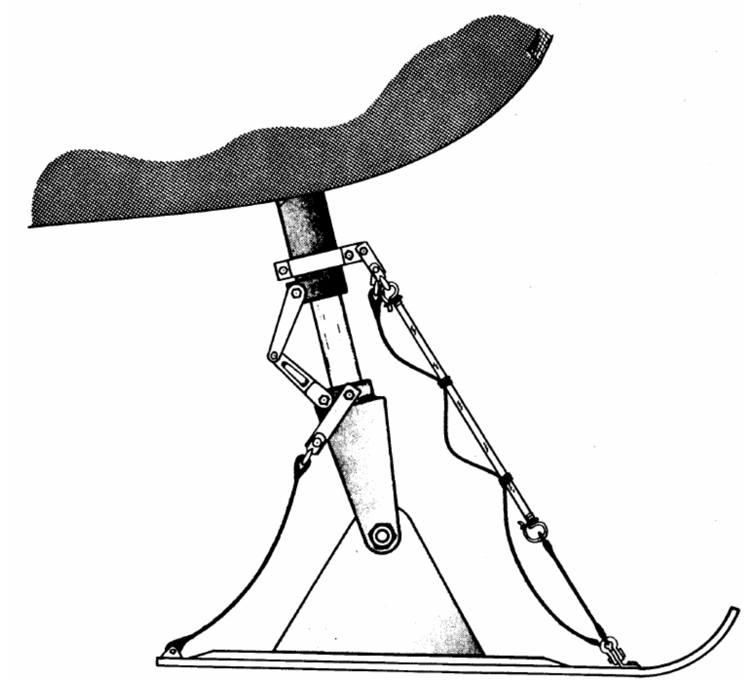
e.

f. Nose Ski Installation. The nose ski is installed in the same manner as the main skis (see Figure 5-7), except:

(1) Adjust length of safety cable to provide -5- to -15-degree ski incidence.

(2) Where it is possible for the nose ski rigging to contact the propeller tips due to vibration, install a 1/4-inch shock cord to hold the rigging out of the propeller arc.

FIGURE 5-7. TYPICAL NOSE SKI INSTALLATION



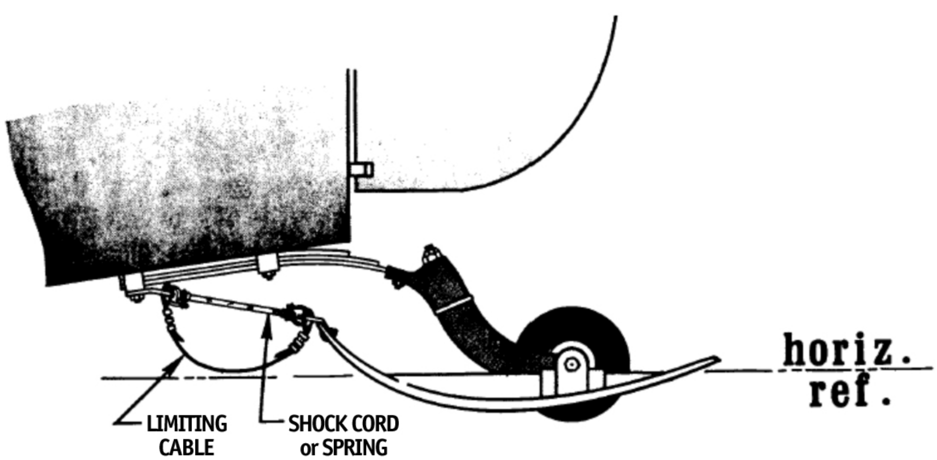
g. Tail Ski Installation.

(1) Use tail skis that have been approved on airplanes of approximately the same weight (within 10 percent) or select as outlined in paragraph 2, Hazards and Warnings. Depending upon the type of ski selected, the tail wheel may or may not have to be removed.

(2) Adjust the length of the limiting cable (reference Figure 5-8) to allow the ski to turn approximately 35 degrees either side of the straight-forward position with the weight of the airplane resting on the ski.

(3) The shock cord (reference Figure 5-8) must be of a length that will hold the ski in the straight-forward position during flight.

FIGURE 5-8. TYPICAL TAIL SKI INSTALLATION



506. DOCUMENTATION.

a. Manufacturer Data. Comply with the applicable requirements for placards or manuals as listed in the approved or accepted documents discussed in paragraph 4, Installation Considerations.

b. Performance Information. The following FAA policies contain the minimum additional performance data and operational check flights for ski installations. AC 43-210, chapter 4, as amended, should be consulted for additional guidance on Aircraft Flight Manual Supplements (AFMS).

(1) For aircraft over 6,000 pounds maximum certificated weight, obtain FAA approval for an AFMS that adds the following or similar performance section information.

(a) Takeoff. Under the most favorable conditions of smoothly packed snow at temperatures approximating 32° F, the ski plane takeoff distance is approximately 10 percent greater than that shown for the landplane.

NOTE: In estimating takeoff distance for other conditions, caution should be exercised as lower temperatures or other snow conditions will usually increase these distances.

(b) Landing. Under the most favorable conditions of smoothly packed snow at temperatures approximately 32° F, the ski plane landing distance is approximately 20 percent greater than that shown for the landplane.

NOTE: In estimating landing distances for other conditions, caution should be exercised as other temperatures or other snow conditions may either decrease or increase these distances.

(c) Climb Performance. In cases where the landing gear is fixed (both landplane and ski plane), where climb requirements are not critical, and the climb reduction is small (30 to 50 feet per minute), the FAA will accept a statement of the approximate reduction in climb performance placed in the performance information section of the AFMS. For larger variations in climb performance, where the minimum requirements are critical, or the landing gear of the landplane is retractable, appropriate climb data should be obtained to determine the changes and new curves, tables, or a note should be incorporated into the AFMS.

(2) For aircraft of 6,000 pounds or less maximum certificated weight, make the information in paragraph 5-5-2a available to the pilot in the form of placards, markings, manuals, or any combination thereof.

507. FLIGHT AND HANDLING OPERATIONAL CHECKS. Accomplish an operational check in accordance with Title 14 of the Code of Federal Regulations (14 CFR) part 91, § 91.407 that includes more than one landing to determine the ground-handling characteristics as well as takeoff and landing characteristics. Take note of ski angles during tail high and tail low landings to avoid having the ski dig in or fail from localized stress. Determine if there is sufficient ground control to satisfactorily complete a landing run with a turnoff at slow speed in cases where brakes are not provided. In flight, the ski should ride steady with no unusual drag and produce no unsatisfactory flight characteristics. Enter a notation of these checks in the aircraft records.

508. MAINTENANCE (INCLUDING INSPECTION).

a. Inspection and Repair Data Sources. Contact the aircraft and ski manufacturer for any specific inspection and maintenance instructions they may have developed. Refer to AC 43.13-1, chapter 9, as amended, for more information.

b. Instructions for Continued Airworthiness (ICA). The developer of the alteration (ski installation or ski modification) must provide instructions for future inspection and maintenance of the added or altered parts, and is also responsible for assessing the need for any changes to the product-level ICA (changes that affect the aircraft as a whole). For simple aircraft/ski combinations where the skis were previously approved on another similar aircraft with appropriate ICA data, it may only be necessary to reference the data in the newly altered aircraft maintenance records and/or FAA Form 337, block 8. For complex installations with special considerations, specific ICA may need to be developed by the modifier. In either case, the modifier must ensure that the adequate and appropriate ICA is available to the owner or operator. Consult AC 43-210 and FAA Order 8110.54, (as amended) for additional guidance on ICA.

c. Interchanging of Skis and Wheels. A person appropriately authorized by 14 CFR part 43, § 43.3 must perform the initial ski installation's new weight-and-balance computation. We recommend that the aircraft be weighed for this initial computation. After the initial installation, removing the skis and reinstalling the wheels or vice versa may be considered a preventive maintenance operation when no weight-and-balance computation or complex assembly is involved.

NOTE: During subsequent weight-and-balance changes to the aircraft, make sure all landing gear configuration, weight-and-balance records (skis, wheels, floats), equipment list are updated.

d. Periodic Inspection Required. Seasonally removed and installed equipment such as skis should be inspected at installation to comply with § 91.409, if they were not installed on the aircraft at the time of the last inspection. All available data described in this section should be used during the inspection.

509. THRU 599. RESERVED

CHAPTER 6. OXYGEN SYSTEM INSTALLATIONS IN NONPRESSURIZED AIRCRAFT

SECTION 1. GENERAL

600. PURPOSE. This chapter provides data for acceptable means of gaseous oxygen system installations in non-pressurized aircraft. For other oxygen system installations (i.e., liquid oxygen), installer should contact their local Flight Standard District Office (FSDO) for assistance in applying for a Supplemental Type Certificate (STC).

601. HAZARDS/WARNINGS TO CONSIDER WHEN INSTALLING AN OXYGEN SYSTEM.

a. Oxygen itself does not burn, but materials that burn in air will burn much hotter and more vigorously in a oxygen rich environment.

b. Oil and Grease burn with explosive violence in the presence of oxygen.

c. Rapid release of high-pressure oxygen in the presence of foreign particles can cause temperatures sufficient to ignite combustible materials and materials that would not normally burn in air.

d. Pressurized oxygen cylinders failures, particularly aluminum-lined composite cylinders, have the potential of producing violent explosions.

e. Flexible hoses are not recommended for use in high-pressure oxygen equipment when pressure exceeds 500 psi.

602. ADDITIONAL REFERENCES.

a. Title 14 of the Code of Federal Regulations (14 CFR) part 23, Airworthiness Standards: Normal, Utility, Acrobatic, and Commuter Category Airplanes.

b. Part 27, Airworthiness Standards: Normal Category Rotorcraft.

c. Part 43, Maintenance, Preventive Maintenance, Rebuilding, and Alteration.

d. Part 91, General Operating and Flight Rules.

e. Civil Aviation Regulations (CAR) 3, Airplane Airworthiness; Normal, Utility, and Acrobatic Categories.

f. CAR 6, Rotorcraft Airworthiness; Normal Category.

g. Advisory Circular (AC) 27-1, Certification of Normal Category Rotorcraft.

h. AC 43.13-1, Acceptable Methods, Techniques, and Practices – Aircraft Inspection and Repair.

i. Society of Automotive Engineers Aerospace Information Report (SAE AIR) No. 825B, Oxygen Equipment for Aircraft.

j. Handbook Bulletin for Airworthiness (HBAW) 02-01B, Maintenance of Pressure Cylinders in Use as Aircraft Equipment.

k. FAA Order 8310.6 Airworthiness Compliance Check Sheet Handbook.

603. THRU 606. RESERVED

SECTION 2. INSTALLATION OF THE OXYGEN SYSTEM

607. SYSTEM REQUIREMENTS.

a. Cylinders. Install oxygen cylinders confirming to Interstate Commerce Commission (ICC) requirements for gas cylinders which carry the ICC or DOT 3A, 3AA, or 3HT designation followed by the service pressure metal-stamped on the cylinder.

b. Tubing/Lines.

(1) In systems having low pressure (400 psi), use seamless aluminum alloy or equivalent having an outside diameter of 5/16 inch and a wall thickness of .035". Double flare the ends to attach to fittings.

(2) In high-pressure systems (1800 psi), use 3/16 inch O.D., .035" wall thickness, seamless copper alloy tubing meeting Specification WWT-779a type N, or stainless steel between the filler valve and the pressure-reducing valve. Silver-solder cone nipples to the ends of the tubing to attach the fittings in accordance with Specification MIL-B-7883.

(3) Use 5/16-inch O.D. aluminum alloy tubing after the pressure-reducer (low-pressure side).

NOTE: Any lines that pass through potential fire zones should be stainless steel.

(4) Use flexible connections specifically designed for oxygen between all points having relative or differential motion.

NOTE: If lines are located behind upholstery or not 100% visible during normal operations, they should be solid metal lines or high-pressure flexible lines.

c. Fittings. All fittings must be manufactured from materials that are compatible for use with oxygen systems. Fittings should not be made of mild steel or materials that are prone to corrosion when in contact with another material.

(1) High Pressure. Intercylinder connections are made with regular flared or flareless tube fittings with stainless steel. Usually fittings are of the same material as the lines. Mild steel or aluminum alloy fittings with stainless steel lines are discouraged. Titanium fittings should never be used because of a possible chemical reaction and resulting fire.

(2) Low Pressure. Fittings for metallic low-pressure lines are flared or flareless, similar to high pressure lines. Line assemblies should be terminated with "B" nuts in a similarly manner to a manufactured terminating connection. Universal adapters (AN 807) or friction nipples used in conjunction with hose clamps are not acceptable for use in pressurized oxygen systems.

d. Valves. Each system must contain a slow-opening/closing shutoff valve that is assessable to a flight crewmember to turn on and shut off the oxygen supply at the high pressure source.

e. Regulators. The cylinder or system pressure is reduced to the individual cabin outlets by means of a pressure-reducing regulator that can be manually or automatically controlled. The regulator should be mounted as close as possible to the cylinders and certificated for aviation environment.

f. Types of Regulators. The four basic types of oxygen systems, classified according to the type of regulator employed, are:

(1) Continuous-flow type. The constant flow type provides the same output pressure or flow regardless of altitude. Continuous-flow oxygen systems provide protection for passengers up to 25,000 feet mean sea level (MSL) and a continuous flow of 100 percent oxygen to the user. They may be automatic or manual in their operation.

(2) Diluter-demand type. This type

dilutes pure oxygen with ambient air and maintains the proper portion of oxygen in the breathing gas depending on altitude. Oxygen concentration is automatically diluted proportionate to the specific altitude's predetermined human oxygen consumption requirement. Such systems only supply oxygen, mixed with cabin air, during inhalation.

(3) Demand type. This type uses high-pressure compressed oxygen that feeds a supply of oxygen to a 'high-to-low' pressure regulator at the individual crew station. The regulator, after reducing the higher pressure, automatically cycled low-pressure breathing oxygen to the wearer only on demand. Section 23.1441 requires aircraft that are certified to operate above 25,000 feet to be equipped with a demand system to supply required crewmembers.

(4) Pressure-demand type. This type is not likely to be included on small general-aviation models. Aircraft certified to exceed 40,000 feet must have a pressure-demand system, which delivers pressurized oxygen to pilots: A High Altitude Oxygen breathing system (above 40,000 feet) using anthropomorphic facial measurements of aircrew to produce a mask that would satisfactorily contain the pressure required to allow breathing under pressure, while maintaining an airtight face-seal and also remain relatively comfortable to the wearer.

g. Flow Indicators.

(1) A pith-ball flow indicator, vane, wheel anemometer, or lateral pressure indicator which fluctuates with changes in flow or any other satisfactory flow indicator may be used in a continuous flow-type system.

(2) An Air Force-Navy flow indicator or equivalent may be used in a diluter-demand type system. Each flow indicator should give positive indication when oxygen flow is occurring.

h. Relief Valve.

(1) A relief valve is installed in low-pressure oxygen systems to safely relieve excessive pressure, such as that caused by overcharging.

(2) A relief valve is installed in high-pressure oxygen systems to safely relieve excessive pressure, such as that caused by heating.

i. Gauge. Provide a pressure gauge to show the amount of oxygen in the cylinder during flight.

j. Masks. Only approved masks designed for the particular system should be used.

608. INSTALLATION AND DESIGN CONSIDERATIONS. Oxygen systems present a hazard. Therefore, follow the precautions and practices listed below:

a. Remove oil, grease (including lip salves, hair oil, etc.), and dirt from hands, clothing, and tools before working with oxygen equipment.

b. Prior to cutting the upholstery, check the intended route of the system.

CAUTION: Ensure all system components are kept completely free of oil or grease during installation and locate components so they will not contact or become contaminated by oil or oil lines.

c. Keep open ends of cleaned and dried tubing capped or plugged at all times, except during attachment or detachment of parts. Do not use tape, rags, or paper.

d. Clean all lines and fittings that have not been cleaned and sealed by one of the following methods:

(1) A vapor-degreasing method with stabilized trichlorethylene conforming to Specification MIL-T-7003 or carbon tetrachloride. Blow tubing clean and dry with a stream of clean, dried, water-pumped air, or dry nitrogen (water-vapor content of less than 0.005 milligrams per liter of gas at 700º F and 760 millimeters of mercury pressure).

(2) Flush with naptha conforming to Specification TT-N-95; blow clean and dry off all solvent with water-pumped air; flush with antiicing fluid conforming to Specification MIL-F-5566 or anhydrous ethyl alcohol; rinse thoroughly with fresh water; and dry thoroughly with a stream of clean, dried, water-pumped air, or by heating at a temperature of 250º to 300º F for one-half hour.

(3) Flush with hot inhibited alkaline cleaner until free from oil and grease; rinse thoroughly with fresh water; and dry thoroughly with a stream of clean, dried, water-pumped air, or by heating at a temperature of 250º to 300º F for one-half hour.

e. Install lines, fittings, and equipment above and at least 6 inches away from fuel, oil, and hydraulic systems. Use deflector plates where necessary to keep hydraulic fluids away from the lines, fittings, and equipment.

f. Allow at least a 2-inch clearance between the plumbing and any flexible control cable or other flexible moving parts of the aircraft. Provide at least a 1/2-inch clearance between the plumbing and any rigid control tubes or other rigid moving parts of the aircraft.

g. Allow a 6-inch separation between the plumbing and the flight and engine control cables, and electrical lines. When electrical conduit is used, this separation between the plumbing and conduit may be reduced to 2 inches.

h. Route the oxygen system tubing, fittings, and equipment away from hot air ducts and equipment. Insulate or provide space between these items to prevent heating the oxygen system.

i. Mount all plumbing in a manner that prevents vibration and chafing. Support a 3/16-inch O.D. copper line each 24 inches and a 3/16-inch O.D. aluminum each 36 inches with cushioned loop-type line support clamps (AN-742) or equivalent.

j. Locate the oxygen supply valve (control valve) so as to allow its operation by the pilot during flight. The cylinder shutoff valve may be used as the supply control valve, if it is operable from the pilot's seat. Manifold plug-in type outlets, which are incorporated in automatic systems, may be considered oxygen supply valves since the pilot can control the flow of oxygen by engaging and disengaging the plug-in type oxygen mask.

k. Filler connections, if provided, are recommended to be located outside the fuselage skin or isolated in a manner that would prevent leaking oxygen form entering the aircraft. Careful evaluation should also be made of any nearby source of fuel, oil, or hydraulic fluid under normal or malfunction conditions. Each filler connection should be placarded. Additionally, any valve (aircraft or ground servicing equipment) associated with high pressure should be slow acting.

NOTE: Locate the oxygen shutoff valve on or as close as practicable to the cylinder to prevent loss of oxygen due to leakage in the system.

609. EQUIPMENT LOCATION AND MOUNTING. Determine the weight/load factors and c.g., limits for the installation prior to commencing the installation.

a. Mount the cylinder in the baggage compartment or other suitable location in such a position that the shutoff valve is readily accessible. Provide access to this valve from inside the cabin so that it may be turned on or off in flight.

b. Fasten the cylinder brackets securely to the aircraft, preferably to a frame member or floorboard using AN bolts with fiber or similar locking nuts. Add sufficient plates, gussets, stringers, cross-bracing, or other reinforcements, where necessary, to provide a mounting that will withstand the inertia forces, stipulated in chapter 1.

c. When cylinders are located where they may be damaged by baggage or stored materials, protect them with a suitable guard or covering.

d. Provide at least 1/2 inch of clear airspace between any cylinder and a firewall or shroud isolating a designated fire zone.

e. Mount the regulator close to the cylinder to avoid long high-pressure lines.

f. Store the masks in such a way that there will be a minimum delay in removing and putting them into use.

610. THREAD COMPOUND. Use anti seize or thread-sealing compound conforming to Specification MIL-T-5542-B, or equivalent.

a. Do not use compound on aluminum alloy flared tube fittings having straight threads. Proper flaring and tightening should be sufficient to make a flared tube connection leakproof.

b. Treat all male-tapered pipe threads with antiseize and sealing compound (MIL-T-5542-B, or tetrafluroethylene tape MIL-T-27730), or equivalent.

c. Apply the compound in accordance with the manufacturer's recommendation. Make sure that the compounds are carefully and sparingly applied only to male threads, coating the first three threads from the end of the fitting. Do not use compound on the coupling sleeves or on the outside of the tube flares.

611. FUNCTIONAL TEST. Before inspection plates, cover plates, or upholstering are replaced, make a system check including at least the following:

a. Open cylinder valve slowly and observe the pressure gauge.

b. Open supply valve and remove one of the mask tubes and bayonet fittings from one of the masks in the kit. Plug the bayonet into each of the oxygen outlets. A small flow should be noted from each of the outlets. This can be detected by holding the tube to the lips while the bayonet is plugged into an outlet.

c. Check the complete system for leaks. This can be done with a soap solution made only from a mild (castile) soap or by leak-detector solution supplied by the oxygen equipment manufacturer.

d. If leaks are found, close the cylinder shutoff valve and reduce the pressure in the system by plugging a mask tube into one of the outlets or by carefully loosening one of the connections in the system. When the pressure has been reduced to zero, make the necessary repairs. Repeat the procedure until no leaks are found in the system.

CAUTION: Never tighten oxygen system fittings with oxygen pressure applied.

e. Test each outlet for leaks at the point where the mask tube plugs in. This can be done by using a soapy solution over each of the outlets. Use the solution sparingly to prevent dogging the outlet by soap. Remove all residue to prevent accumulation of dirt.

f. Examine the system to determine if the flow of oxygen through each outlet is at least equal to the minimum required by the pertinent requirements at all altitudes at which the aircraft is to be operated. This can be accomplished by one of the following methods:

(1) In a continuous flow system when the calibration (inlet pressure vs. flow) of the orifices used at the plug-in outlets is known, the pressure in the low-pressure distribution line can be measured at the point which is subject to the greatest pressure drop. Do this with oxygen flowing from all outlets. The pressure thus measured should indicate a flow equal to or greater than the minimum flow required.

(2) In lieu of the above procedure, the flow of oxygen, through the outlet that is subject to the greatest pressure drop, may be measured with all other outlets open. Gas meters, rotometers, or other suitable means may be used to measure flows.

(3) The measurement of oxygen flow in a continuous flow system which uses a manually adjusted regulator can be accomplished at sea level. However, in a continuous flow system which uses an automatic-type regulator, it may be necessary to check the flow at maximum altitude which will be encountered during the normal operation of the aircraft. The manufacturer of the particular continuous-flow regulator used should be able to furnish data on the operating characteristics of the regulator from which it can be determined if a flight check is necessary.

(4) Checking the amount of flow through the various outlets in a diluter-demand or straight-demand system is not necessary since the flow characteristics of the particular regulator being used may be obtained from the manufacturer of the regulator. However, in such systems the availability of oxygen to each regulator should be checked by turning the lever of the diluter-demand regulator to the "100 percent oxygen" position and inhaling through the tube via the mask to determine whether the regulator valve and the flow indicator are operating.

g. Provide one of the following acceptable means or equivalent to indicate oxygen flow to each user by:

(1) Listening for audible indication of oxygen flow.

(2) Watching for inflation of the rebreather or reservoir bag.

(3) Installation of a flow indicator.

612. OPERATING INSTRUCTIONS. Provide instructions appropriate to the type of system and masks installed for the pilot on placards. Include in these instructions a graph or a table which will show the duration of the oxygen supply for the various cylinder pressures and pressure altitudes (Table 6-1).

613. THRU 617. RESERVED

TABLE 6-1. TYPICAL OXYGEN DURATION TABLE

ACTUAL DURATION IN HOURS AT VARIOUS ALTITUDES					
Number of Persons	*8000 Ft.*	*10,000 Ft.*	*12,000 Ft.*	*15,000 Ft.*	*20,000 Ft.*
Pilot only	7.6 hr	7.1 hr	6.7 hr	6.35 hr	5.83 hr
Pilot and 1 Passenger	5.07 hr	4.74 hr	4.47 hr	4.24 hr	3.88 hr
Pilot and 2 Passengers	3.8 hr	3.55 hr	3.36 hr	3.18 hr	2.92 hr
Pilot and 3 Passengers	3.04 hr	2.84 hr	2.68 hr	2.54 hr	2.34 hr
Pilot and 4 Passengers	2.53 hr	2.37 hr	2.24 hr	2.12 hr	1.94 hr
NOTE: The above duration time is based on a fully charged 48 cubic-foot cylinder. For duration using 63 cubic-foot cylinder, multiply any duration by 1.3.					

FIGURE 6-1. TYPICAL FLOOR MOUNTING

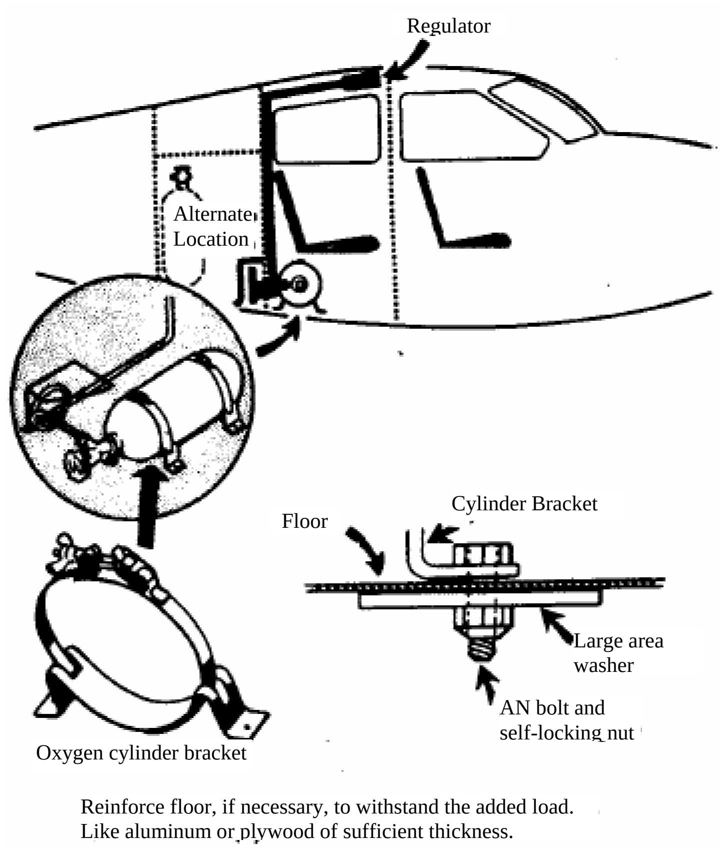


FIGURE 6-2. TYPICAL BAGGAGE COMPARTMENT MOUNTING

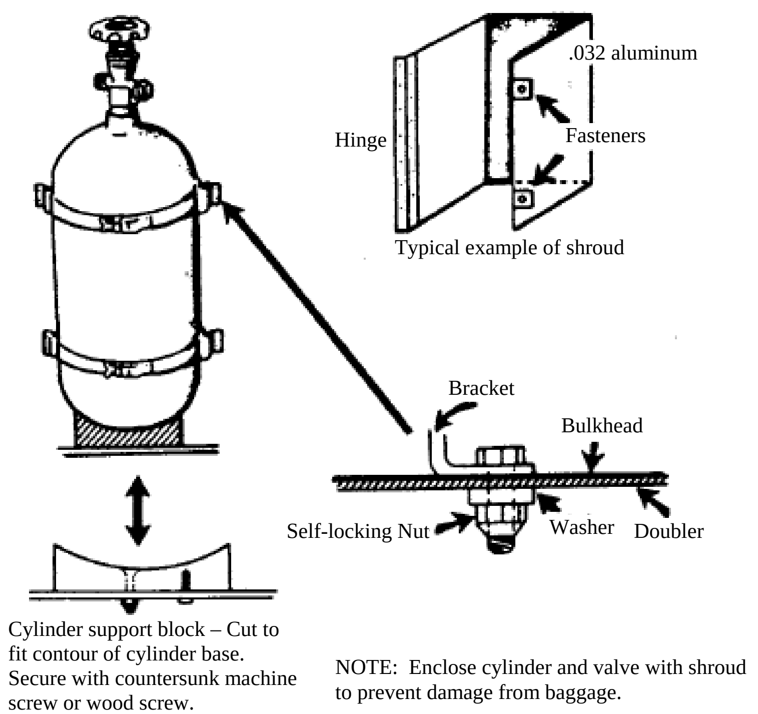
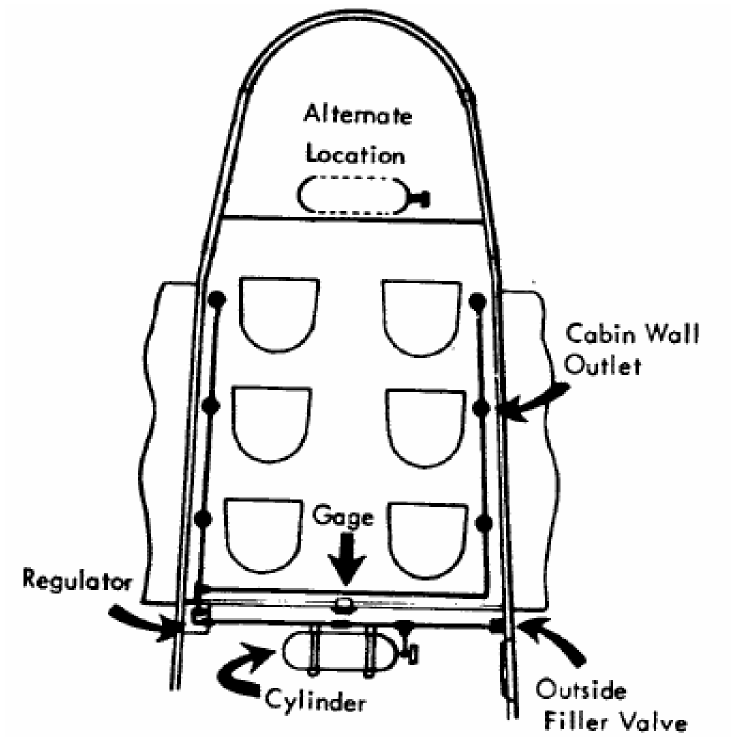


FIGURE 6-3. TYPICAL OXYGEN INSTALLATION IN LIGHT TWIN AIRCRAFT



SECTION 3. AIRWORTHINESS COMPLIANCE CHECK SHEET: OXYGEN SYSTEM INSTALLATION IN UNPRESSURIZED AIRCRAFT

618. GENERAL. Oxygen system installations which are the same as those made by the airframe manufacturer, or other installations which are already approved, may be accepted without further investigation. On other installations, the following points should be checked to determine if the installation is satisfactory.

619. APPLICABLE FEDERAL AVIATION REGULATIONS. Does the installation comply with the regulations listed in section 1, paragraph 602, Additional References?

620. STRUCTURAL REQUIREMENTS.

a. If changes or alterations of the aircraft structure are made (mounting of oxygen cylinder), is the original strength and integrity of the structure retained?

b. If gages are added to the instrument panel or an outside filler valve installed on the fuselage, is the structural integrity of the panel and airframe or its supporting structure retained?

c. Has the extent the modification affected the center of gravity (c.g.) of the aircraft evaluated?

d. Are all lines properly routed and supported?

621. HAZARDS TO THE AIRCRAFT OR ITS OCCUPANTS. Is the design of the Oxygen system evaluated to ensure that the aircraft and its occupants are safe from hazards identified in section 1, paragraph 601, Hazards/Warnings to Consider when Installing an Oxygen System?

622. OPERATING ASPECTS.

a. When required by the operating rules for the use of supplemental oxygen, is the system capacity sufficient to supply oxygen to all combinations of crew and passengers?

b. Does the oxygen system provide the required flow?

c. Are the oxygen regulator controls accessible to a member of the flightcrew in flight?

d. Is there a means, readily available to the crew in flight, to turn on and shut off the oxygen supply at the high pressure source?

e. Is there a means to allow the crew to readily determine, during flight, the quantity of oxygen available in each source of supply?

623. DETAIL DESIGN STANDARDS.

a. Are all parts suitable for use with oxygen?

b. If flexible hoses are used in a high pressure oxygen system, are only certified oxygen-compatible polytetrafluorethylene (PTFE) lined flexible hoses used?

c. In high pressure oxygen applications, is the use of polymer flexible hoses kept to a minimum and used in conjunction with solid tubing with the same internal diameter as the hose at the downstream end before any restriction (valve, sharp bend, reduction in size of hose or tube, etc.)?

d. Is the regulator located as close as physically possible to the Oxygen cylinder and minimizes the use of fittings?

e. Are any lines not 100 percent visible during normal operation solid metal lines?

f. Does each breathing device have a device attached that visually shows the flow of Oxygen?

g. Are the lines that pass through potential fire zones stainless steel?

h. When oxygen system components are added to the baggage compartments, are there provisions to protect the system components from shifting cargo?

i. Where oxygen components are installed, is the compartment placarded against the storage of oil or hydrocarbons?

j. Is a smoke detector installed where oxygen cylinders are installed in a closed, nonaccessible compartment?

k. Is the cargo area weight limitations placard updated?

l. Flexible hoses are not recommended for use in high pressure Oxygen equipment when pressure exceeds 500 psi unless special precautions are taken, or the outcome for fire is possible. Are these precautions taken?

m. Are all Oxygen outlets placarded?

n. Are "No Smoking When Oxygen Is In Use" placards and other appropriate placards (i.e., operation instructions) visible to the crew and occupants of the aircraft?

624. INSTRUCTIONS FOR CONTINUED AIRWORTHINESS.

a. Are there written instructions concerning system operation, maintenance, and cylinder changing procedures?

b. Are there written instructions in the use of the oxygen equipment in the aircraft flight manual or placards?

NOTE: Any changes to the aircraft's flight manual must be approved by the FAA.

c. Are there written inspection and test schedules and procedures?

d. Are mechanical drawings and wiring diagrams available, as required?

e. Are there written instructions concerning oxygen cylinder hydrostatic testing requirements and cylinder replacement requirements?

f. Is there a scheduled (annual) check of a constant flow system manifold output pressure for recommended output pressure?

g. Is the guidance contained in AC 43.13-1B, paragraphs 9-47 to 9-51 taken into consideration in the design of the oxygen maintenance program?

625. RECORDKEEPING

a. Has a maintenance record entry been made? (Reference § 43.9.)

b. Has the equipment list and weight and balance been revised? (Reference Order 8310.6 Airworthiness Compliance Check Sheet Handbook, chapter 1.)

626. THRU 699. RESERVED

CHAPTER 7. ROTORCRAFT EXTERNAL-LOAD-DEVICE INSTALLATIONS CARGO SLINGS AND EXTERNAL RACKS

SECTION 1. GENERAL

700. PURPOSE. This section contains structural and design information for the fabrication and installation of a cargo sling used as an external load attaching means for a Class B rotorcraft-load combination operation under Title 14 of the Code of Federal Regulations (14 CFR) part 133. As an external-load attaching means, a "cargo sling" includes a quick-release device (hook) and the associated cables, fittings, etc., used for the attachment of the cargo sling to the rotorcraft. Part 133, § 133.43(d) specifies the requirements for the quick-release device.

701. HAZARDS/WARNINGS. Particular attention should be paid to the effect of the sling load on the lateral as well as the fore and aft center of gravity (c.g.) of the rotorcraft.

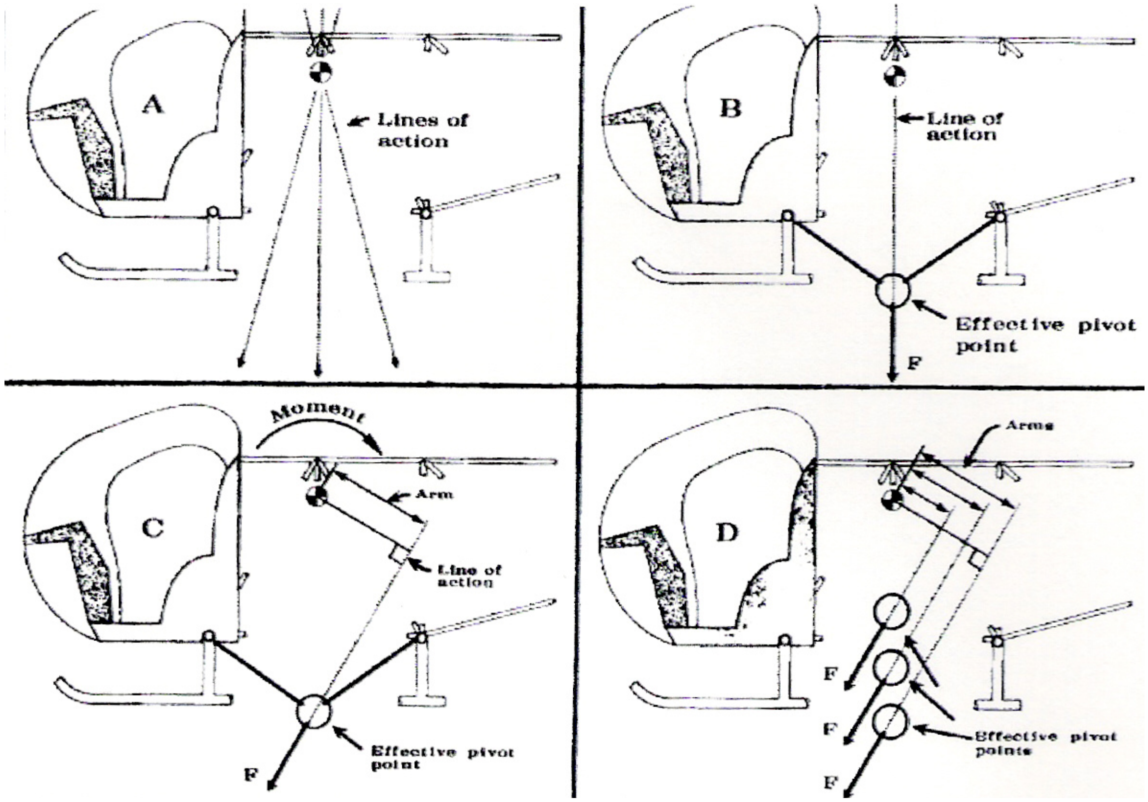
702. REFERENCES. It is the installer's responsibility to ensure that the latest revisions of any regulations, advisory circulars, manufacturers data, etc., are used in any cargo sling or rack installation.

703. INSTALLATION CONSIDERATIONS.

a. Location of Cargo Release in Relation to the Rotor Craft's Center of Gravity Limits. An ideal location would be one that allows the line of action to pass through the rotorcraft's center of gravity at all times. (See Figure 7-1, Illustration A.) However, with most cargo sling installations, this ideal situation is realized only when the line of action is vertical or near vertical and through the rotorcraft's c.g. (See Figure 7-1, Illustration B.)

(1) Whenever the line of action does not pass through the rotorcraft's c.g. due to the attachment method used, acceleration forces, or aerodynamic forces, the rotorcraft-load combined c.g. will shift from the rotorcraft's c.g. position. Depending upon the factors involved, the shift may occur along either or both the longitudinal or lateral axes. The amount of shift is dependent upon the force applied (F) and the length of the arm of the line of action. Their product (F x Arm) yields a moment which can be used to determine the rotorcraft-load combined c.g.. (See Figure 7-1, Illustration C.) If the rotorcraft-load c.g. is allowed to shift beyond the rotorcraft's approved c.g. limits, the rotorcraft may become violently uncontrollable.

FIGURE 7-1. LOCATION OF CARGO RELEASE IN RELATION TO THE ROTORCRAFT'S CENTER OF GRAVITY



(2) Thus, any attachment method or location which will decrease the length of the arm will reduce the distance that the combined c.g. will shift for a given load (F) and line of action angle. (See Figure 7-1, Illustration D.)

b. Maximum External Load. The maximum external load (including the weight of the cargo sling) for which authorization is requested may not exceed the rated capacity of the quick-release device.

704. FABRICATION AND INSTALLATION.

a. Static Test. The cargo sling installation must be able to withstand the static load required by §133.43(a). Conduct the test as outlined in chapter 1. If required during the test, supports may be placed at the landing gear to airframe attach fittings to prevent detrimental deformation of the landing gear due to the weight of the aircraft.

b. Sling-Leg Angles of Cable-Supported Slings. The optimum sling-leg angle (measured from the horizontal) is 45 to 60 degrees. Minimum tension in a sling leg occurs with a sling-leg angle of 90 degrees, and the tension approaches infinity as the angle approaches zero. Thus, larger sling-leg angles are desirable from a standpoint of cable strength requirements. Slings should not be attached in such a manner as to provide sling-leg angles of less than 30 degrees.

c. Minimum Sling-Leg Cable Strength.

(1) An analysis which considered the effects of 30-degree sling angles showed that the minimum cable strength design factor required would be 2.5 times the maximum external load for each leg regardless of the number of legs. Although this is the minimum strength required by part 133, it may be desirable to double this value to allow for deterioration of the sling-leg cables in service. This will result in a cable strength equal to five times the maximum external load.

Example: Maximum external load 850 pounds
Minimum required sling-leg cable
strength 850 x 2.5 = 2125
Minimum desired sling-leg cable
strength 850 x 2.5 x 2 = 4250

(2) A 3/16-inch, nonflexible 19-wire cable (MIL-W-6940*)* provides a satisfactory cable strength. See AC 43.13-1A, chapter 4, Figure 4-1, for a table of breaking strength of steel cable. For convenience, the cable sizes desired for various loads have been calculated and are tabulated in Table 7-1 based on a factor of 5:

TABLE 7-1. CABLE LOAD TABLE

Maximum External Load (pounds)	Aircraft Cable Size For Each Cargo Sling Leg		
	MIL-C-5693 and MIL-W-6940	MIL-W-1511	MIL-C-5424
100	1/16	3/32	3/32
200	3/32	1/8	1/8
300	7/64	1/8	1/8
400	1/8	1/8	5/32
500	5/32	5/32	3/16
600	5/32	3/16	3/16
700	3/16	3/16	3/16
800	3/16	3/16	7/32
900	3/16	7/32	7/32
1,000	7/32	7/32	7/32
1,200	7/32	1/4	¼
1,400	¼	1/4	9/32
1,600	1/4	9/32	5/16
1,800	5/16	5/16	5/16
2,000	5/16	11/32	3/8

d. Sling Installation.

(1) Attach the cargo sling to landing gear members or other structure capable of supporting the loads to be carried. Install the quick-release device in a level attitude with the throat opening facing the direction as indicated on the quick-release device. When cables are used to support the quick-release device, make sure the cables are not twisted or allowed to twist in the direction to un-lay the cable.

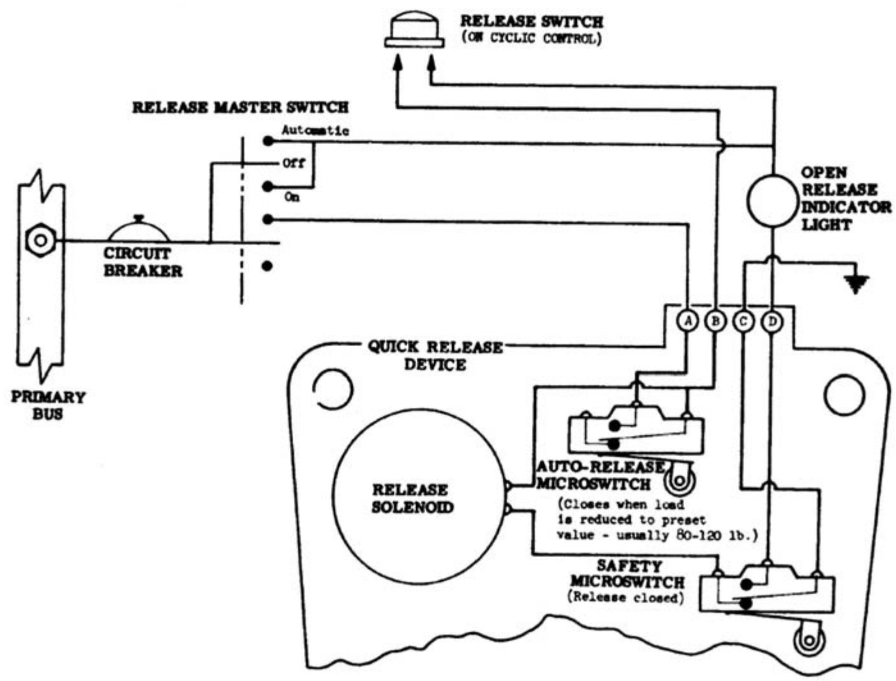
(2) Some cargo release devices are provided with a fitting to permit installation of a guideline to assist in fully automatic engagement of the load target ring or load bridle. Secure the guideline to the quick-release device with a shear pin of a definite known value which will shear if a load becomes entangled on or over the guideline. Provision should also be made for cable-supported slings to be drawn up against the fuselage into a stowage position to prevent striking or dragging the release on the ground when not in use.

e. Installation of Release Controls. See Figure 7-2 for typical wiring diagram of the electrical controls.

(1) Install a cargo release master switch, readily accessible to the pilot, to provide a means of deactivating the release circuit. The power for the electrical release circuit should originate at the primary bus. The "auto" position of the release master switch on some cargo release units provides for automatic release when the load contacts the ground and the load on the release is reduced to a preset value.

(2) Install the cargo release switch on one of the pilot's primary controls. It is usually installed on the cyclic stick to allow the pilot to release the load with minimum distraction after maneuvering the load into the release position.

FIGURE 7-2. TYPICAL CARGO SLING WIRING DIAGRAM



(3) Install the emergency manual release control in a suitable position that is readily accessible to the pilot or other crewmember. Allow sufficient slack in the control cable to permit complete cargo movement without tripping the cargo release.

(4) The manual ground release handle, a feature of some cargo release units, permits opening of the cargo release by ground personnel.

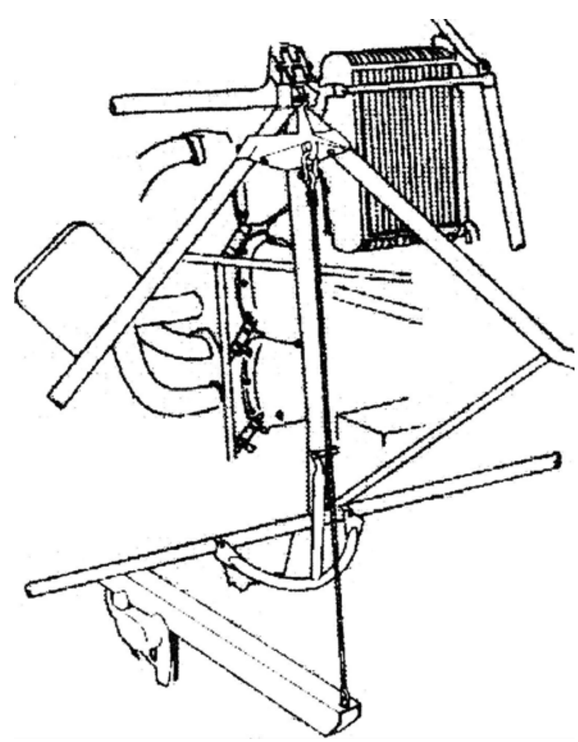
(5) Label or placard all release controls as to each function and operation.

f. Functional Test.

(1) Test the release action of each release control of the quick-release device with various loads up to and including the maximum external load. This may be done in a test fixture or while installed on the rotorcraft, if the necessary load can be applied.

(2) If the quick-release device incorporates an automatic release, the unit should not release the load when the master switch is placed in the "automatic" position until the load on the device is reduced to the preset value, usually 80 to 120 pounds.

FIGURE 7-3. TYPICAL CARGO SLING INSTALLATION NO. 1



g. Supplemental Flight Information. The aircraft may not be used in part 133 external-load operations until a Rotorcraft-Load Combination Flight Manual is prepared in accordance with § 133.47 of that part. Appropriate entries should also be made in the aircraft's weight and balance data, equipment list, and logbooks. The FAA Form 337 should also be executed as required by part 43, § 43.5(a) and (b).

h. Inspection and Maintenance. Inspection of the complete installation should be accomplished prior to each lift for security and functionality. Maintenance should be accomplished in accordance with the manufacturers instructions.

705. THRU 706. RESERVED

FIGURE 7-4. TYPICAL CARGO SLING INSTALLATION NO. 1
(SHOWING FUSELAGE ATTACHMENT FITTING)

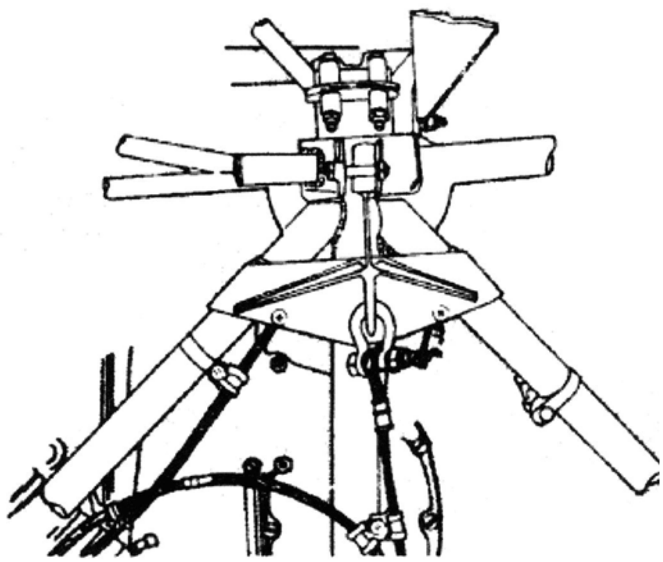


FIGURE 7-5. TYPICAL CARGO SLING INSTALLATION NO. 1
(SHOWING FORE AFT LIMITING STOPS)

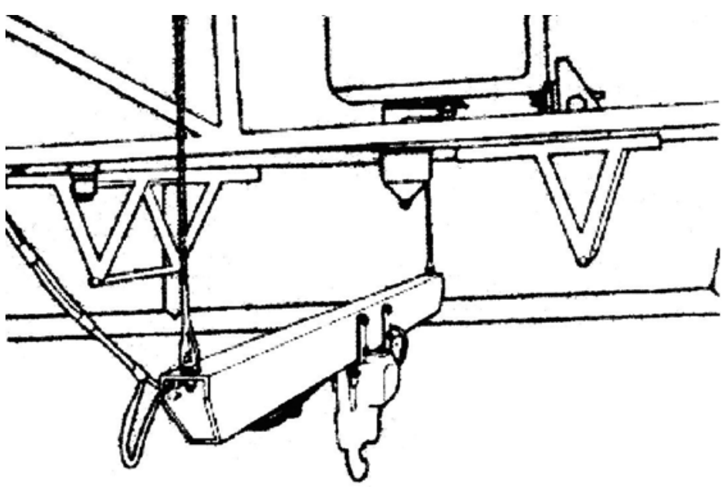


FIGURE 7-6. TYPICAL CARGO SLING INSTALLATION NO. 2 (CARGO HOOK ATTACHED DIRECTLY TO UNDERSIDE OF FUSELAGE)

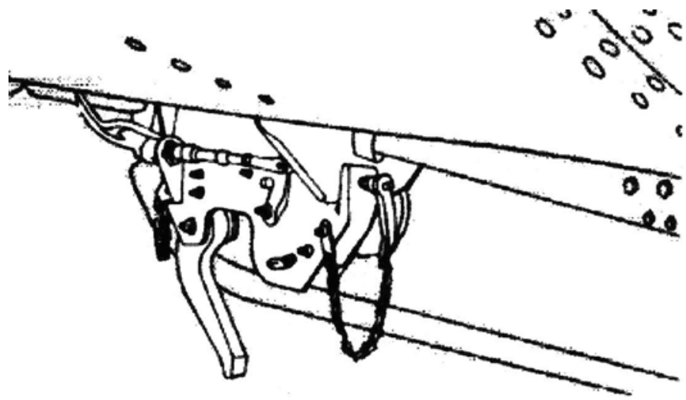


FIGURE 7-7. TYPICAL CARGO SLING INSTALLATION NO. 3 (4-LEG, CABLE SUSPENDED)

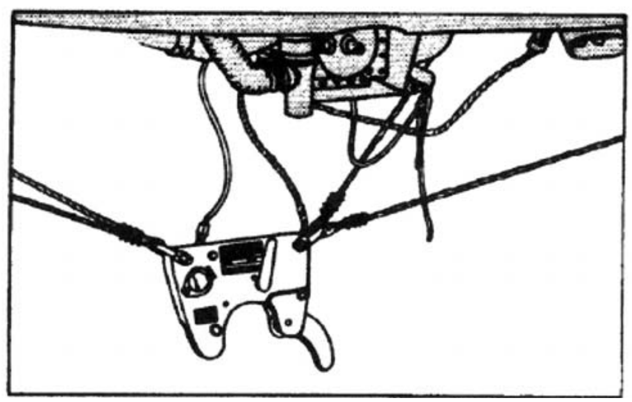


FIGURE 7-8. TYPICAL CARGO SLING INSTALLATION NO. 3 (SHOWING CABLE SLING LEG ATTACHMENT TO LANDING GEAR CROSSTUBE FITTING)

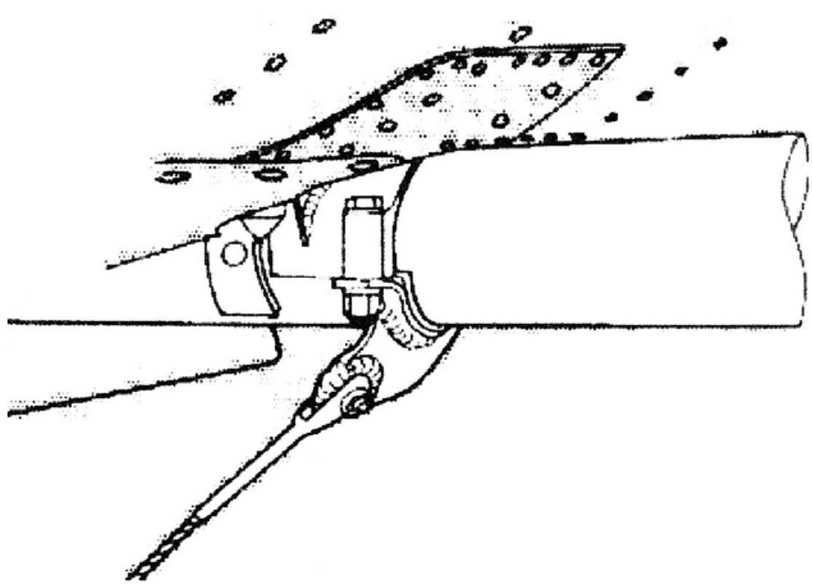
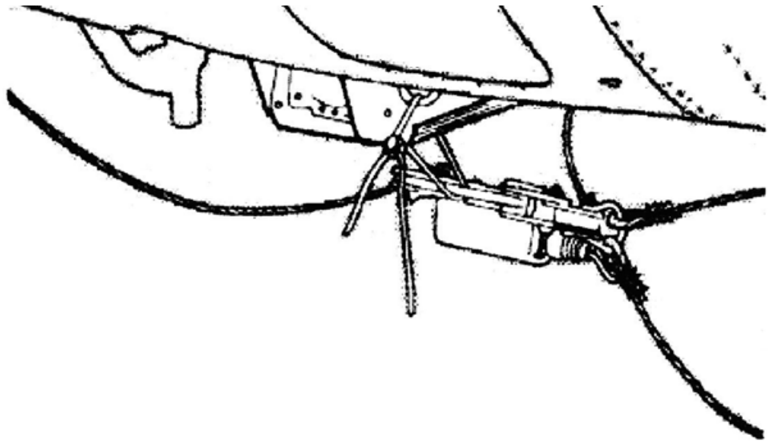


FIGURE 7-9. TYPICAL CARGO SLING INSTALLATION NO. 3 (SHOWING CARGO SLING IN STOWED POSITION)

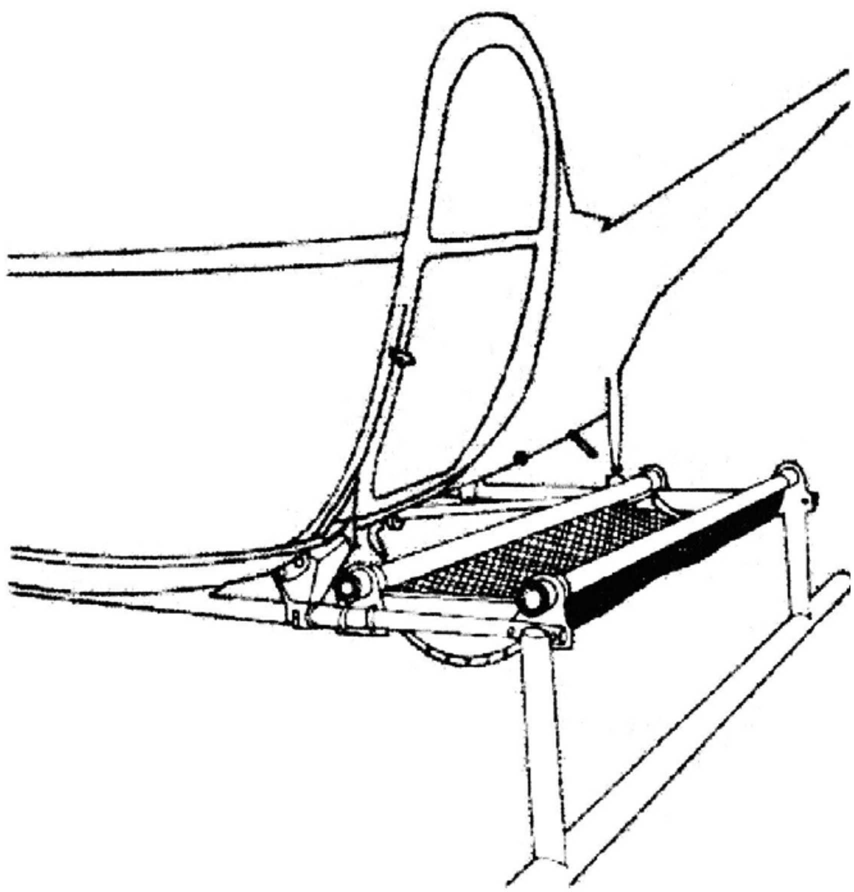


SECTION 2. CARGO RACKS

707. GENERAL. This section contains structural and design information for the fabrication and installation of a cargo rack used as an external-load attaching means for a Class A rotorcraft-load combination operation under part 133.

708. FABRICATION OF CARGO RACKS. The type of construction and method of attachment depends upon the material to be used and the configuration of the rotorcraft involved. Illustrations of typical construction and installation methods are shown in Figures 7-10 through 7-14.

FIGURE 7-10. TYPICAL CARGO RACK INSTALLATION NO. 1



709. STATIC TEST. The cargo rack installation must be able to withstand the static test load required by § 133.43 (a). Conduct the test as outlined in chapter 1.

710. SUPPLEMENTAL FLIGHT INFORMATION. The aircraft may not be used in part 133 external-load operations until a rotorcraft-load combination flight manual is prepared in accordance with § 133.47.

711. THRU 799 RESERVED

FIGURE 7-11. TYPICAL CARGO RACK INSTALLATION NO. 1
(SHOWING ATTACHMENT DETAIL)

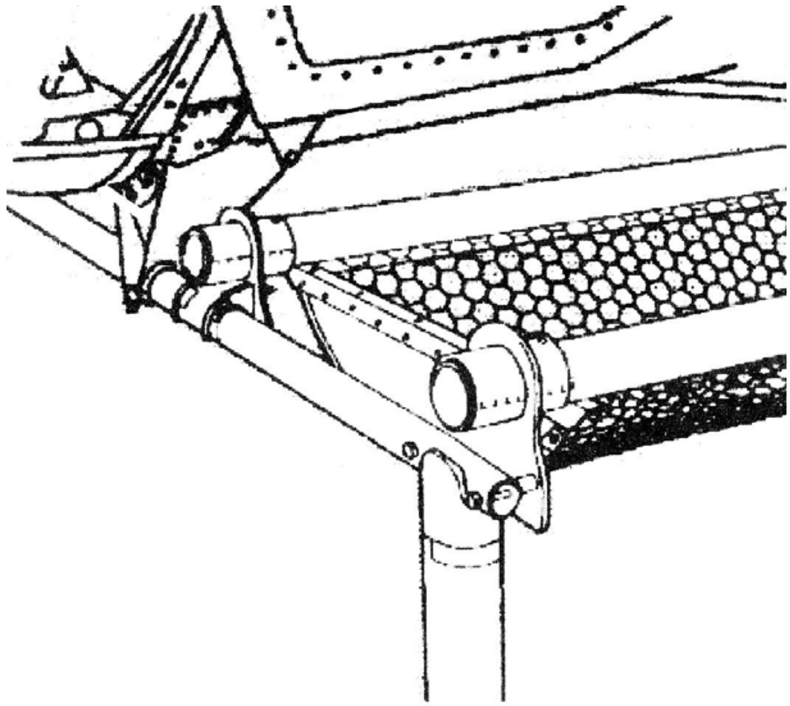


FIGURE 7-12. TYPICAL CARGO RACK INSTALLATION NO. 2

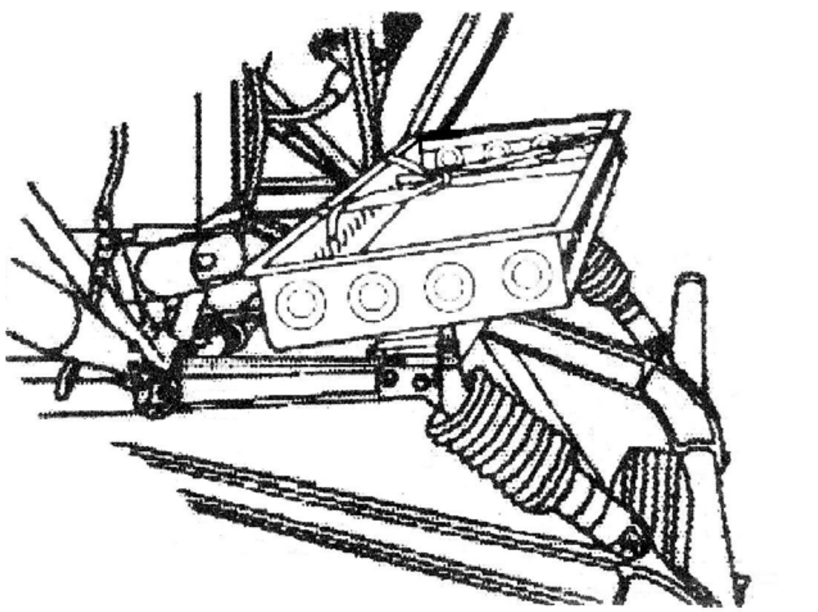


FIGURE 7-13. TYPICAL CARGO RACK INSTALLATION NO. 2
(SHOWING RACK PARTIALLY INSTALLED)

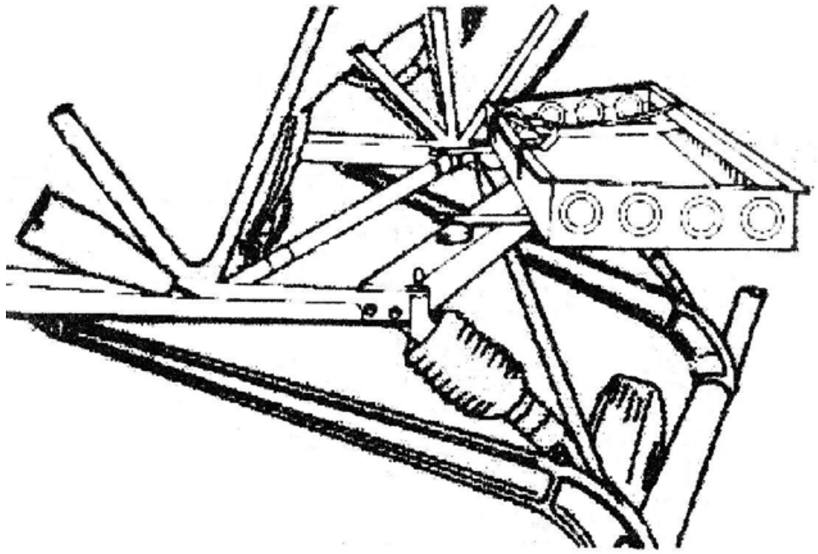
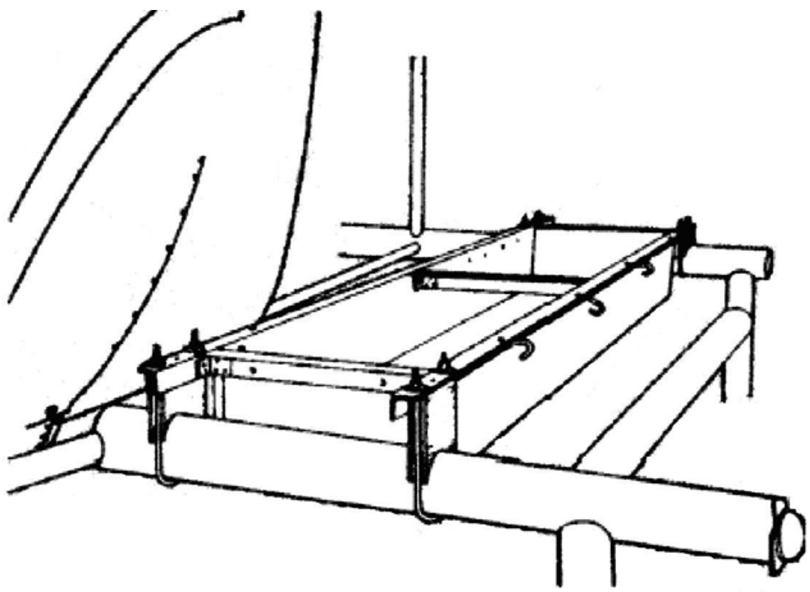


FIGURE 7-14. TYPICAL CARGO RACK INSTALLATION NO. 3



CHAPTER 8. GLIDER AND BANNER TOW-HITCH INSTALLATIONS

800. PURPOSE. This chapter contains design and installation information for banner and glider tow hitches. Guidance for inspection, service, and continuous airworthiness requirements for the hitches are also addressed in this chapter.

801. WARNING. The direction and maximum arc of displacement of banner towline loads occur within a more limited rearward cone of displacement than do glider towline loads (Figure 8-1 and 8-2). Hitches that meet the banner tow criteria of this chapter may not be suitable for glider towing. Due to the basic aerodynamic differences between the two objects being towed, glider and banner tow-hitch installations are treated separately with regard to loading angles.

802. REFERENCES. Occasional reference will be made in this chapter to manufacturer's documents, certain parts of Title 14 of the Code of Federal Regulations (14 CFR), and certain FAA advisory circulars (AC). It is the installer's responsibility to ensure the latest revisions of these documents are used as reference material.

803. INSTALLATION CONSIDERATIONS.

a. Weight and Balance. In most cases, the weight of the tow-hitch assembly will affect the aft center of gravity (c.g.) location. To assure that the possibility of an adverse effect caused by the installation has not been ignored, enter all pertinent computations in the aircraft weight and balance records (in accordance with the provisions contained in 14 CFR part 43, § 43.5(c)(4)). The requirements of § 43.5 (a) and (b) should also be addressed, for maintenance record entries and repair and alteration forms.

b. Equipment List. The aircraft equipment list should be updated to reflect any tow equipment installations. Consideration should also be given to adding appropriate revisions to the aircraft's Pilot Operating Handbook (POH), or Flight Manual, as required.

c. Corrosion Protection. Tow hitches are traditionally simple mechanical devices; however, improper care may lead to hazardous conditions for the aircraft operator. The aft, external location of tow hitches exposes them to the elements, and proper corrosion protection methods should be employed to prevent improper operation or malfunction. Reference should be made to the manufacturers documents or AC 43.13-1, Acceptable Method, techniques, and Practices—Aircraft Inspection and Repair, chapter 6 (current edition) for some additional corrosion protective measures that may be used.

804. FABRICATION AND INSTALLATION PROCEDURES.

a. Methods. Installation of tow hitches should be accomplished using manufacturers data or that data previously approved by a Representative of the Administrator when available, such as a Supplemental Type Certificate or a Field Approval. Installations requiring fabrication of brackets, parts, fittings, etc. should be accomplished using data in the form of ACs such as AC 43.13-1 (current edition), to determine material requirements, load requirements, and type and size of hardware used.

b. Structural Requirements. The structural integrity of a tow-hitch installation on an aircraft is dependent upon its intended usage.

c. Attachment Points. Tow-hitch mechanisms are characteristically attached to, or at, tie down points or tail wheel brackets on the airframe. These are points where the design load-bearing qualities may be sufficient for towing loads. Keep the length from the airframe attachment point to the tow hook at a minimum as the loads on the attachment bolts are multiplied by increases in the attachment arm.

d. Glider Tow Hitch Load Testing.

(1) Protection for the tow-plane is provided by requiring use of a towline assembly that will break prior to structural damage occurring to the tow plane. The normal tow load imposed on the hitch rarely exceeds 80 percent of the weight of the glider. Therefore, the towline assembly design load for a 1,000-pound glider could be estimated at 800 pounds. By multiplying the estimated design load by 1.5 (to provide a safety margin), we arrive at a limit load value of 1,200 pounds. The 1,200-pound limit load value is used in static testing or analysis procedures per paragraph 8-2 to prove the strength of the tow hook installation. When the hook and its attachment to the aircraft structure have been proven to withstand the limit load, 1,200 pounds in this example, then the "maximum" breaking strength of the towline assembly is established at the design load of 800 pounds. Thus, the towline will break well before structural damage will occur to the tow-plane.

FIGURE 8-1. GLIDER TOW ANGLE

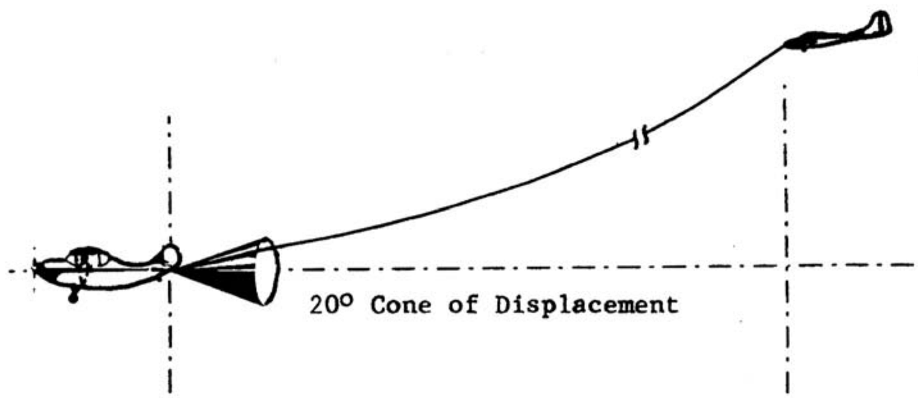
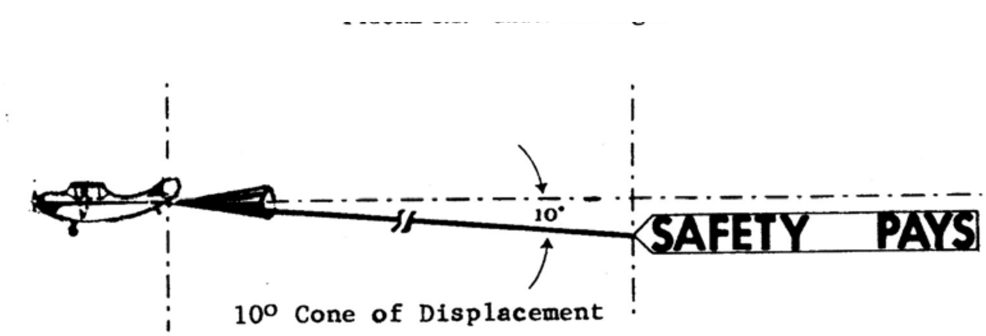


FIGURE 8-2. BANNER TOW ANGLE



(2) Another approach can be applied if the limit load carrying capabilities of a tow hook and fuselage are known. In this case, the known load value can be divided by 1.5 to arrive at the design load limits. For example, if the tow hook and fuselage limit loads are known to be 1,800 pounds, divide this by 1.5 (1800 ÷ 1.5 = 1,200) and we arrive at a design load value of 1,200 pounds. Thus, if a towline assembly rated at 1,200 pounds is used it will break before reaching the structural limits of the hook and attaching structure.

(3) When considering tow hook installations, one may establish maximum towline breaking strength by:

(a) Dividing the known limit load capabilities of the fuselage and tow hook installation by 1.5; or

(b) Knowing the design load needs of the towline assembly and multiplying by 1.5 to arrive at a limit load. Then analysis or static testing, determine that the hook and fuselage are capable of towing the load.

e. Banner Tow Hitches. Install and test the hitch to support a limit load equal to at least two times the operating weight of the banner.

f. Multiple Hitches. Multiple tow hitches are sometimes used on banner tow aircraft. These installations should be evaluated individually for approval.

805. STRUCTURAL TESTING. When installations are made on an aircraft or using brackets that have not been previously approved, some structural testing may be required. Adequacy of the aircraft structure to withstand the required loads can be determined by either static test or structural analysis.

a. Static Testing. When using static tests to verify structural strength, subject the tow hitch to the anticipated limit load. Aircraft used only for banner towing need only be tested to 2 times the weight of the heaviest banner to be towed, using the cone angle shown in Figure 8-2. Aircraft used for glider towing should be tested to a load approximately 80 percent of the weight of the heaviest glider to be towed, using the cone angle shown in Figure 8-1. Testing to be done in accordance with the procedures in chapter 1, paragraph 106.

b. Structural Analysis. If the local fuselage structure is not substantiated by static test for the proposed tow load, using a method that experience has shown to be reliable, subject the fuselage to engineering analysis to determine that the local structure is adequate. Use a fitting factor of 1.15 or greater in the loads for this analysis.

806. ANGLES OF TOW. Tests should be conducted on the system at various tow angles to insure that:

a. There is no interference with the tail wheel or adjacent structure. Tow-hitches mounted to tail wheel springs or trusses, as in Figures 8-6 and 8-7, which travel up and down with the tail wheel, should be tested under load to ensure they don't contact any control surfaces.

b. The towline clears all fixed and movable surfaces at the maximum lateral and vertical cone of displacement and full surface travel.

c. The tow hitch does not swivel. Experience has shown swiveling could result in fouling both the release line and towline during operations by the tow plane.

d. The opened jaw of the hitch does not strike any portion of the aircraft.

e. The hitch is able to release under load at all tow angles.

807. PLACARDS. A placard should be installed in a conspicuous place in the cockpit to notify the pilot of the structural design limits of the tow system. The following are examples of placards to be installed:

a. For glider tow "Glider towline assembly breaking strength not to exceed _____*______ pounds."

b. For banner tow "Tow hitch limited to banner maximum weight of _____**_____ pounds."

*Value established per Paragraph 804D.
**Banner hitch limitations are one-half the load applied per Paragraph 804D.

808. INSTALLATION PROCEDURES — TYPES OF HITCHES. The two most common types of tow hitches in use are the Schweizer

1D112-15 (bolt-on) and 1D112-16 (weld-on), or the Tost G-85 bolt-on. Both are installed using similar mounting brackets and release handles, differing mainly in type of construction and method of tow-rope attachment. Some typical installations on different types of airframes are illustrated in Figures 8-4 through 8-13.

a. Schweizer Hitch: Installation and Maintenance. The Schweizer is a simple over center L-hook type with a rubber tension block to preload the release lever (Figure 8-10). It is eligible for installation on several models of Cessna and Piper aircraft by Supplemental Type Certificate, and has been installed on many other aircraft using the Field Approval process. It uses a tow rope with a single round steel ring attaching to the hook. While this hitch is a simple mechanism, proper maintenance and frequent inspection are necessary to ensure proper operation. All initial installations should be proof tested using the procedures outlined in the latest revision of Schweitzer Aircraft Corp. Form F-236. The following additional procedures are recommended as a minimum, and should be performed at each 100 hr/annual inspection, unless otherwise stated.

(1) Inspect the entire tow hook system for loose or worn pivot pins, damaged fasteners, elongated holes, cracks, corrosion, surface damage, excessive wear, deformed parts, frayed release cable, rubber block damage, and freedom of operation. The mounting location of the hitches leaves them exposed to sunlight and the elements. Ozone and heat can have a detrimental effect on the rubber block. Look for excessive hardness of the rubber block as well as a permanent indentation caused by the contact with the hook lever.

(2) Perform a closing check by verifying that a sufficient closing force is required to compress the rubber block with the pivot hook. The pivot hook should apply sufficient locking load against the latch arm after the latch arm is engaged. Verify that the movement of the latch arm toward the release position causes additional compression of the rubber block.

(3) Perform a no-load pull test at the release arm to verify that a load of 4 to 10 pounds of pull is required to release the lever. See SAC Form F-236 for this procedure. If the release load cannot be obtained within the specified range, the rubber block is deformed and should be replaced with a serviceable block.

NOTE: The above procedures, when properly implemented, help to ensure proper operation of the hitch. When the glider under tow operates above a certain angle to the tow plane, the ring may slide upwards on the hook, causing excessive load on the hook, and difficulty in releasing the tow rope ring. The closing check and pull test should be performed with a standard tow ring in the hook.

(4) For a periodic inspection, a Closing Check should be performed during "each" hookup of the tow rope. Tow plane and glider pilots should follow recommended launch and recovery procedures to help prevent this occurrence.

b. TOST Hitch. The TOST hitch is approved under the German Aviation Authority Type certificate number 60.230/1. It is constructed using a steel housing which contains the hook mechanism, surrounded by a steel ring. The tow rope used with the TOST hitch uses two steel rings, an oblong ring, and a round ring, looped through each other. The round ring is attached to the tow hitch. The ring surrounding the hook, along with the double rings on the tow rope, allows ease of tow release at extreme load angles. (See Figure 8-9.) Installation of the TOST hitch should be accomplished using manufacturers brackets where possible, other installations may need additional evaluation. In all cases, installation, adjustment, maintenance, inspection, and overhaul intervals should be accomplished in accordance with the latest revisions of the TOST Installation Manual for Safety Releases and TOST Operating Manual for tow releases.

WARNING NOTE: Because of the

difference in the size of the steel rings, under no circumstances should the Schweizer single ring rope be used with the TOST hook, nor the TOST double ring rope with the Schweizer hook.

809. TOW RELEASE MECHANISM.

a. Release Lever. A placard indicating the direction of operation should be installed to allay the possibility of confusion or inadvertent operation, and the design of the release lever should provide the following:

(1) Convenience in operation.

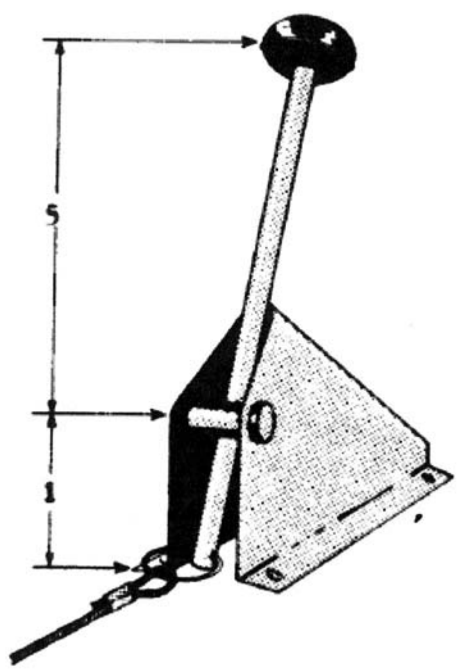
(2) Smooth and positive release operation.

(3) Positioned so as to permit the pilot to easy access without distraction from flight operations, and ease of operation.

(4) Sufficient handle travel to allow for normal slack and stretch of the release cable.

(5) A sufficient handle/lever ratio to assure adequate release force when the tow line is under high loads. (See Figure 8-3.)

FIGURE 8-3. TYPICAL TOW-HITCH RELEASE HANDLE.



(6) Protection of cables from hazards such as:

(a) Wear and abrasion during normal operation.

(b) Binding where cables pass through fairleads, pulleys, etc.

(c) Accidental release.

(d) Interference with or by other aircraft components.

(e) Freezing and moisture accumulation when fixed or flexible tubing guides are used.

b. Test of Release Lever. A test of the release and hook for proper operation through all angles of critical loading should be made using the design load for the glider or banner. The Schwietzer type hitch is susceptible to excessive release loads when the tow cable is at high positive angles, and should be tested before each tow in accordance with the procedures in paragraph 808.

c. Release Cable. Representative size and strength characteristics of steel release cable are as shown in Table 8-1; however, it is recommended that all internally installed release cables be 1/16-inch or larger.

TABLE 8-1. REPRESENTATIVE STEEL CABLE QUALITIES

Diameter inches	Nonflexible Carbon Steel 1 x 7 and 1 x 19 (MIL-W-6904B)		Flexible Carbon Steel 7 x 7 and 7 x 19 (MIL-W-1511A and MIL-C-5424A)	
	Breaking strength (lbs.)	Pounds 100 ft.	Breaking Strength (lbs.)	Pounds 100 ft.
1/32	185	.25	---	---
3/64	375	.55	---	---
1/16	500	.85	480	.75
5/64	800	1.40	---	---
3/32	1,200	2.00	920	1.60

FIGURE 8-4. TRICYCLE GEAR AIRCRAFT

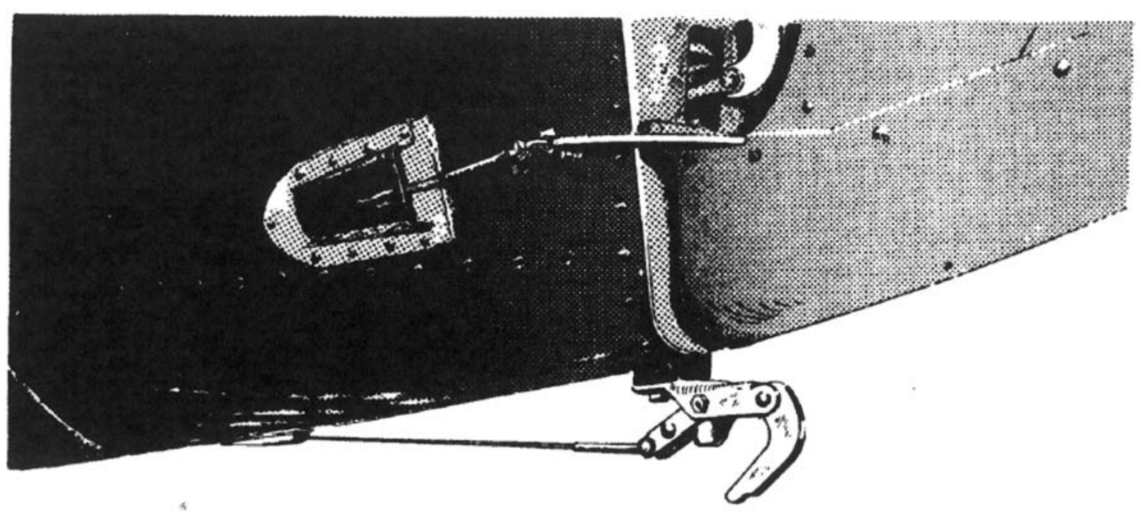


FIGURE 8-5. CONVENTIONAL GEAR AIRCRAFT-LEAF SPRING TYPE TAILWHEEL

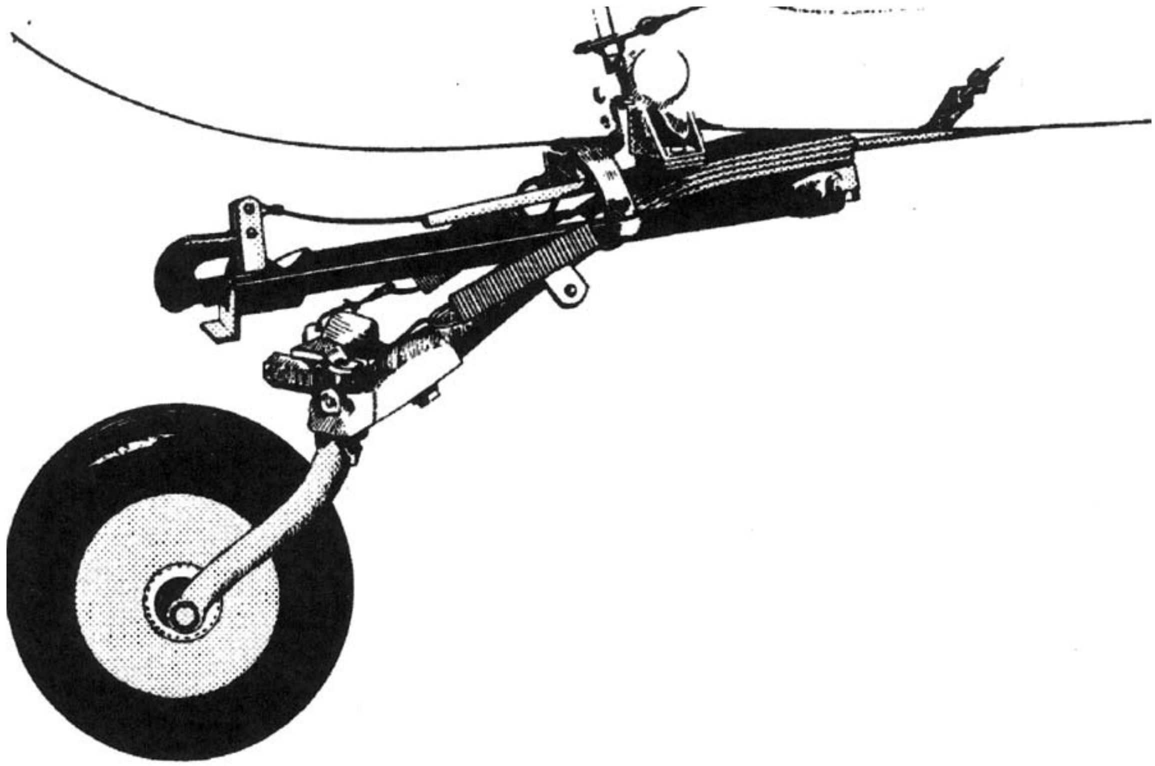


FIGURE 8-6. CONVENTIONAL GEAR AIRCRAFT-SHOCK STRUT TYPE TAILWHEEL

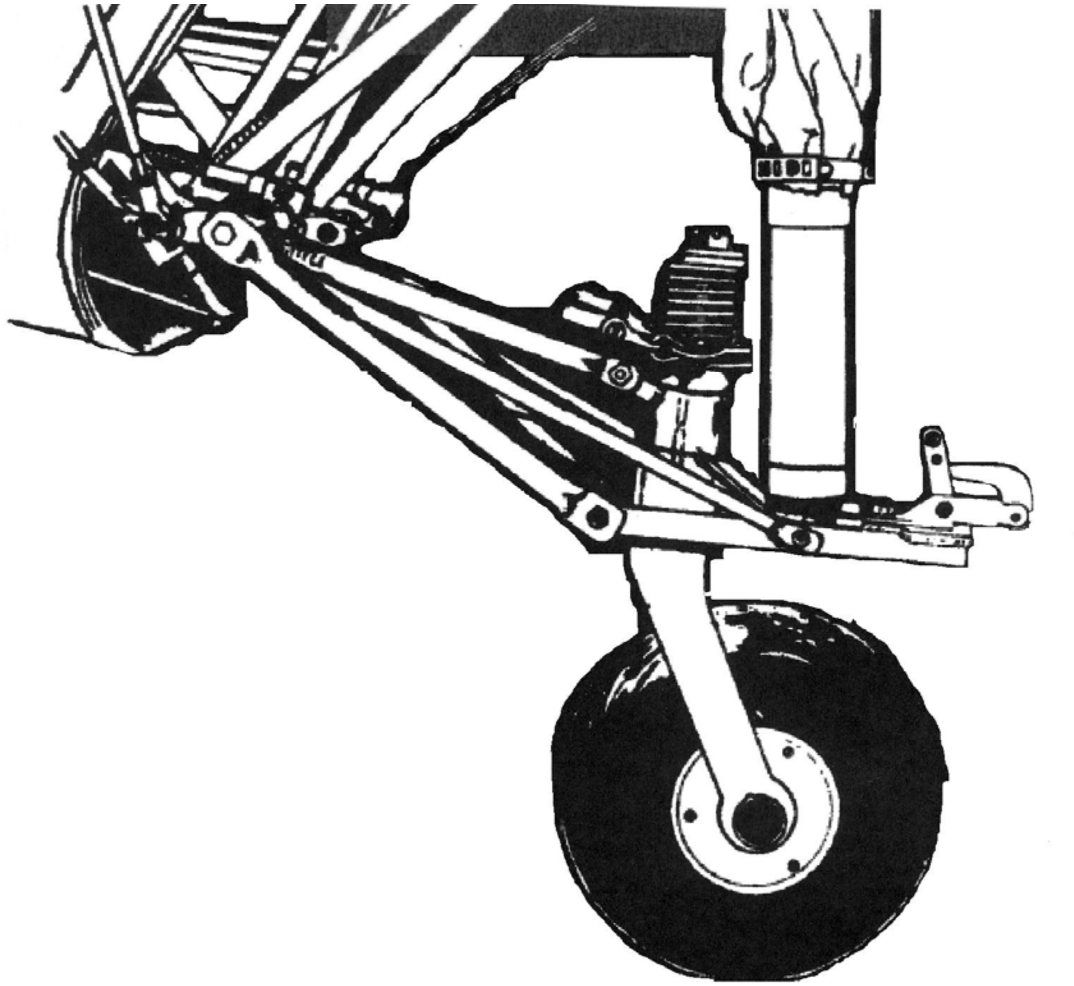


FIGURE 8-7. CONVENTIONAL GEAR AIRCRAFT – TUBULAR SPRING TYPE TAILWHEEL

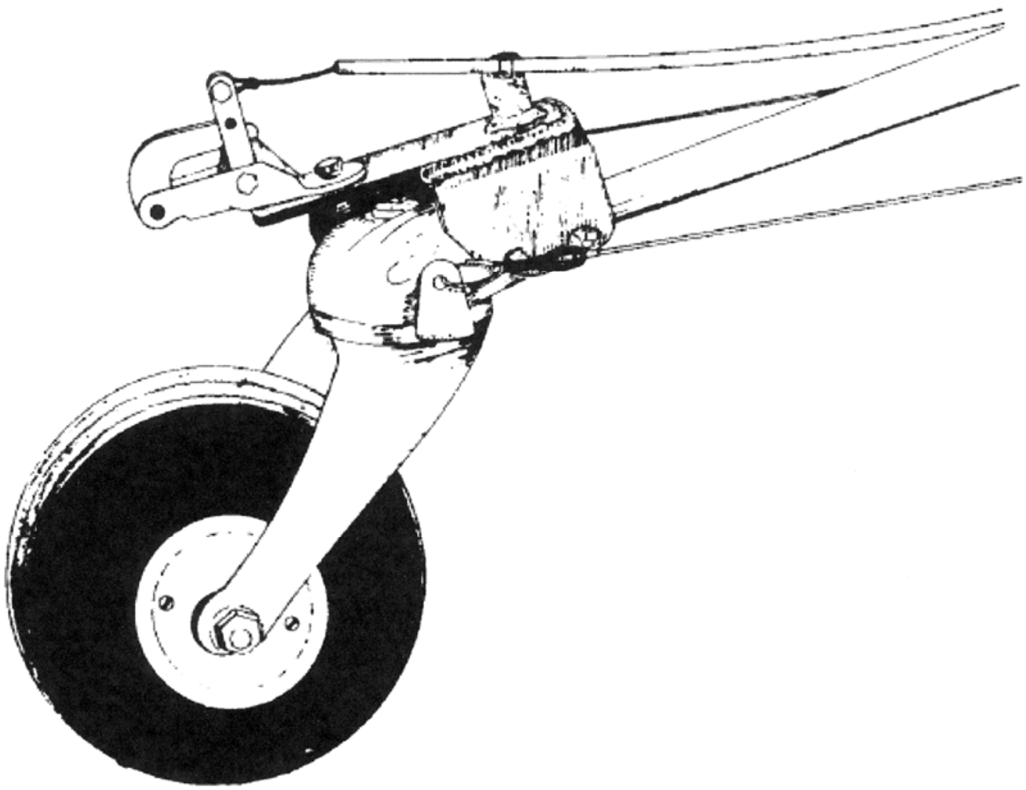


FIGURE 8-8. TYPICAL TOST HITCH

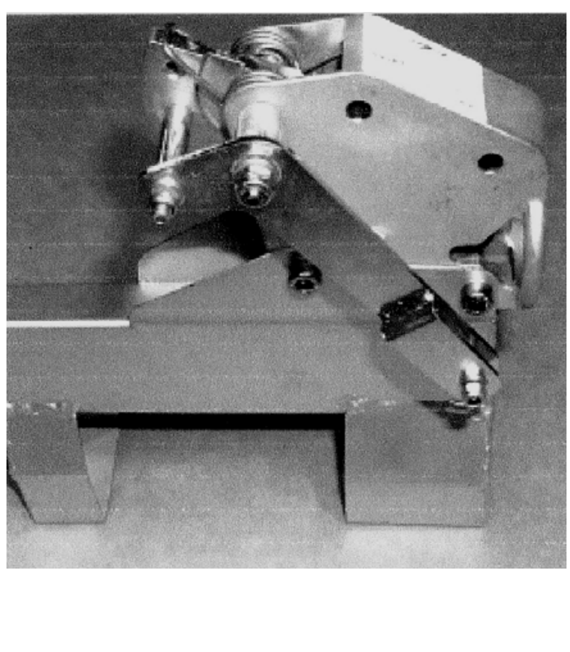


FIGURE 8-9. TOST DOUBLE TOW RINGS

FIGURE 8-10. SCWEIZER WELD ON TYPE HITCH

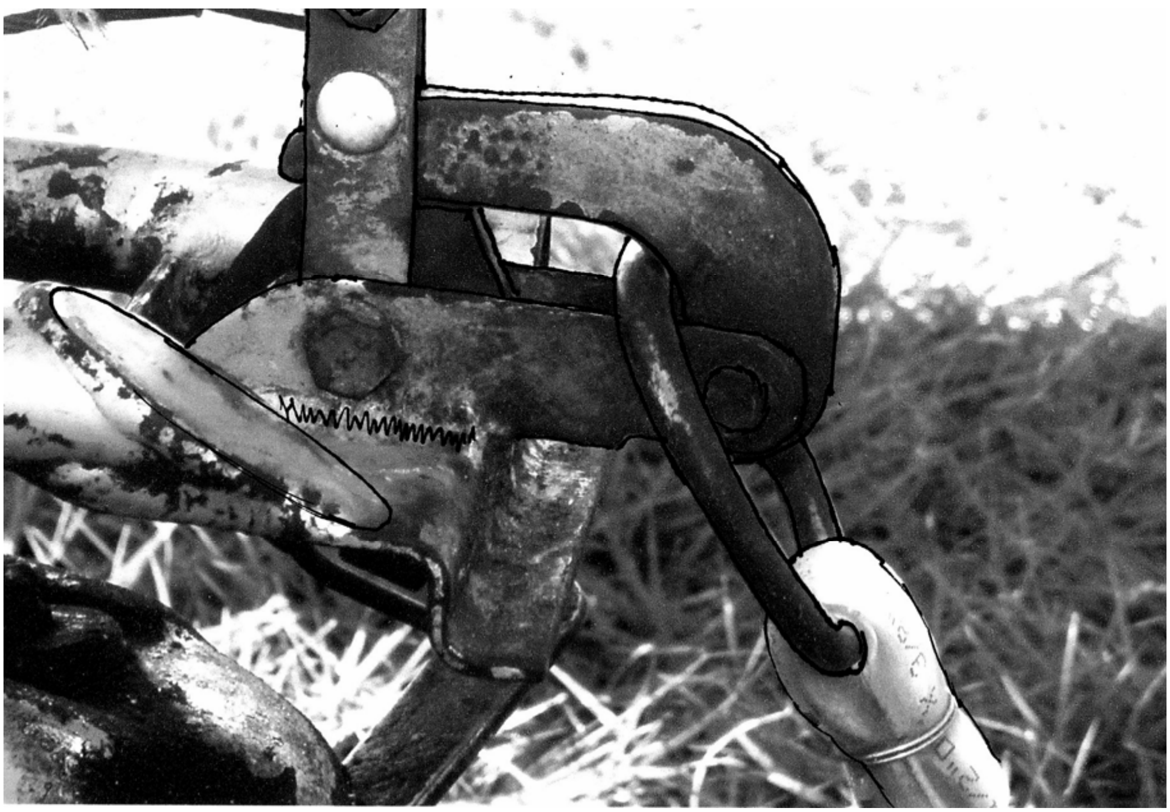


FIGURE 8-11. TYPICAL MULTIPLE HITCH FOR BANNER TOW

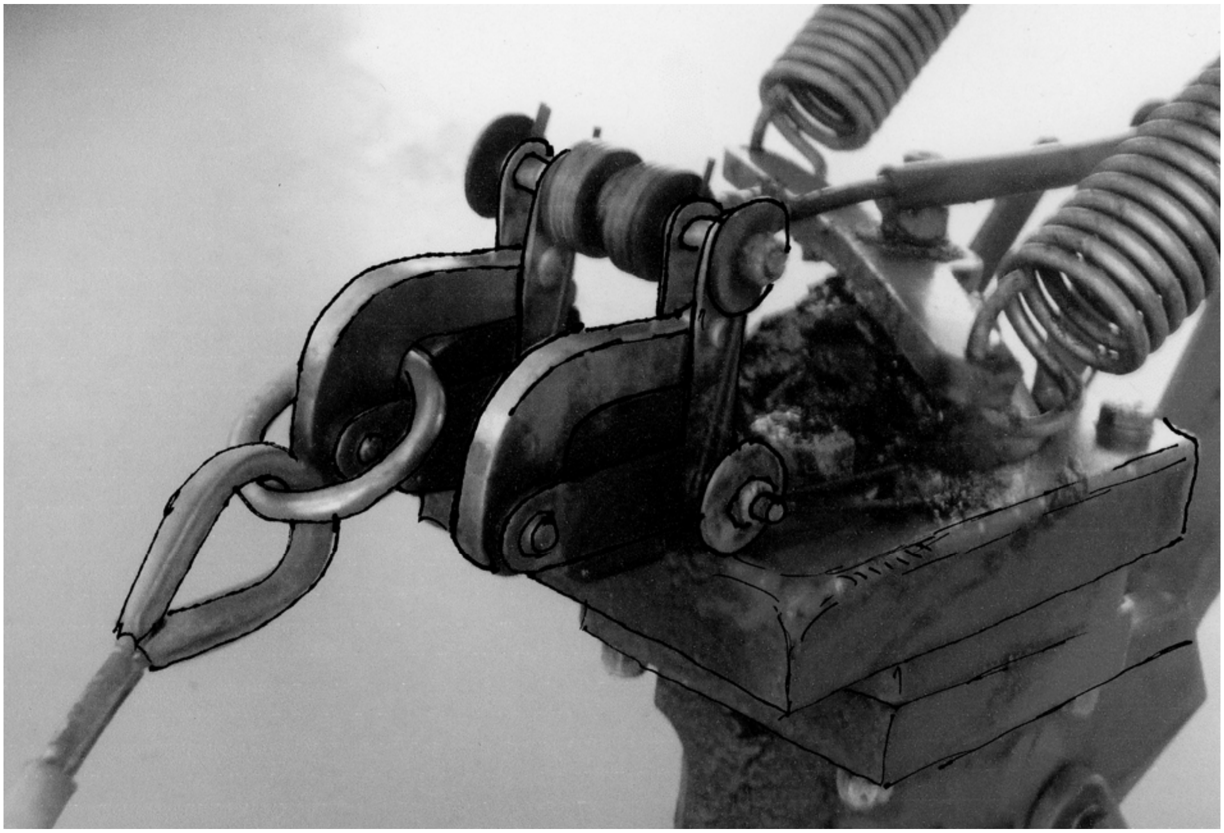


FIGURE 8-12. TYPICAL TOST MOUNT FOR TRICYCLE GEAR AIRCRAFT

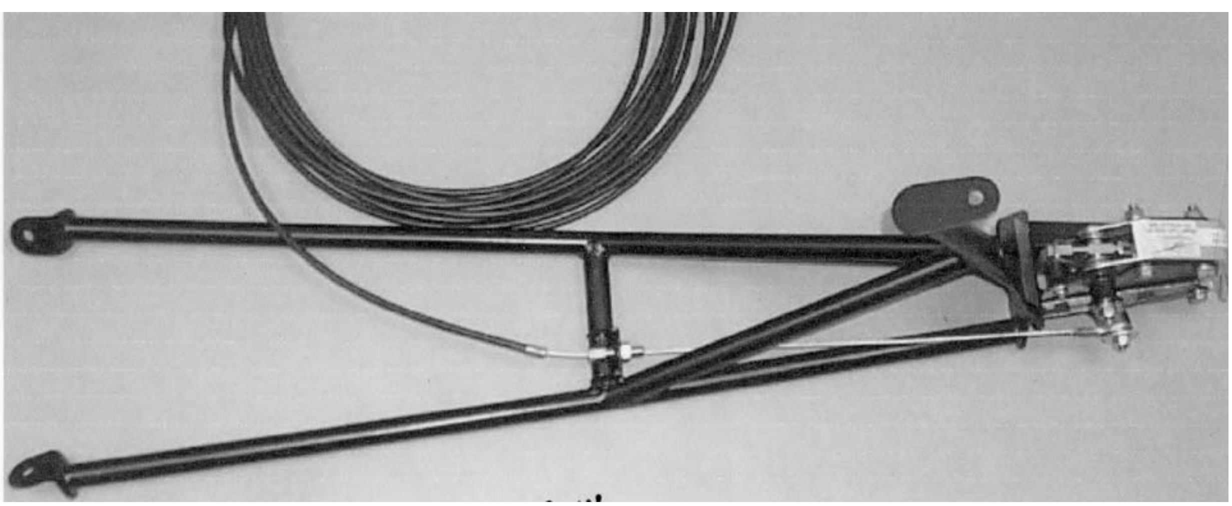
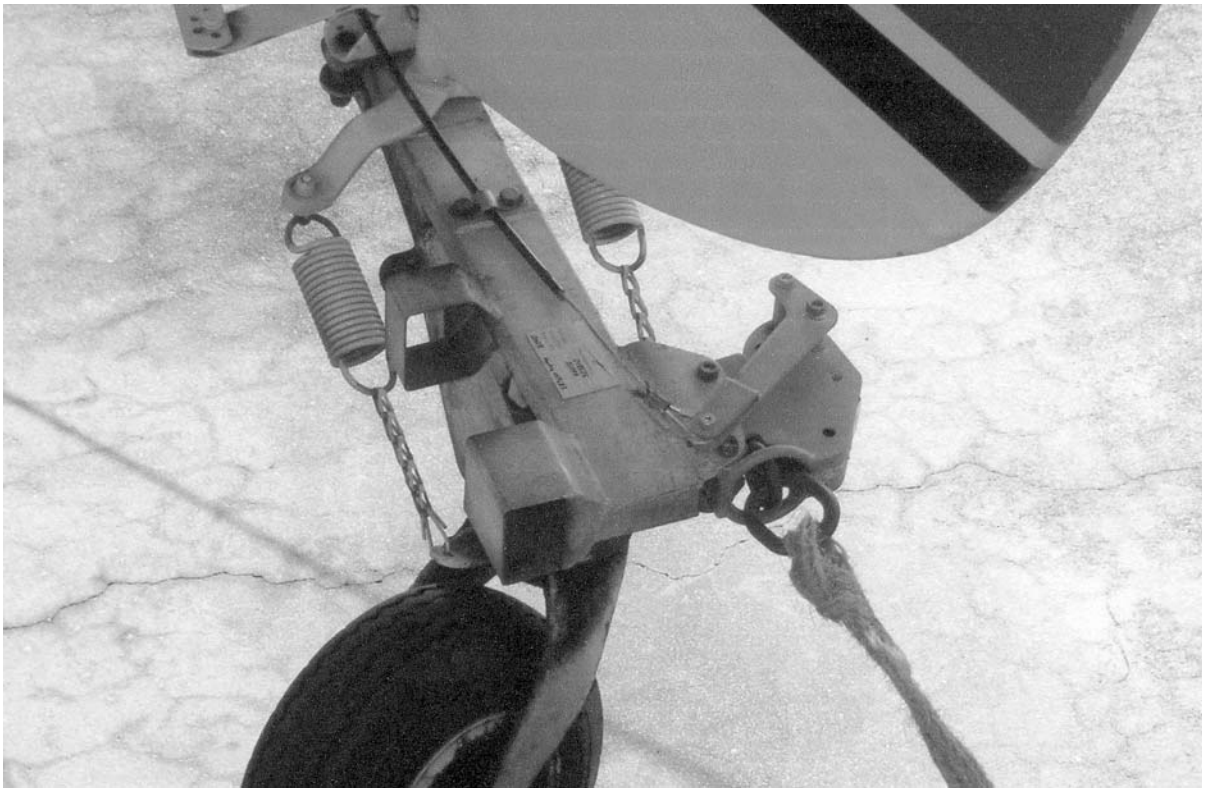


FIGURE 8-13. TYPICAL TOST CONVENTIONAL LEAF SPRING MOUNT



CHAPTER 9. SHOULDER HARNESS INSTALLATIONS

SECTION 1. GENERAL

900. PURPOSE. The purpose of this chapter is to provide guidance for retrofit shoulder harness installations. Information contained herein may be adaptable for the installation of shoulder harnesses in aircraft for which aircraft manufacturer or Supplemental Type Certificate (STC) retrofit installations have not been developed.

901. HAZARDS AND WARNINGS. Installations that do not meet the minimum standards prescribed by regulations are not acceptable. At no time should a retrofit shoulder harness installation perform at less than the static test load requirements specified in section 3, Table 9-1, Static Test Requirements.

902. ADDITIONAL REFERENCES. The following references provide additional information for shoulder harness installations.

a. AC 21-34, Shoulder Harness-Safety Belt Installations.

b. AC 23-17B, Systems and Equipment Guide for Certification of Part 23 Airplanes and Airships.

c. AC 91-65, Use of Shoulder Harnesses in Passenger Seats.

d. TSO C-22G, Safety Belts.

e. TSO –C114, Torso Restraint Systems.

f. Aerospace Standard SAE, AS8043, Restraint Systems for Civil Aircraft.

903. INSTALLATION METHODS. Shoulder harness installations can be performed by minor or by major alterations to the type design depending on the complexity.

a. Minor alterations are limited to those where no change in the aircraft structure is required for mounting the harness. If the installation does not require operations such as drilling holes into or welding brackets onto the primary structure, it could be classified as a minor alteration. (See Figure 9-1.) Two examples of minor alterations for shoulder harnesses are:

(1) Some aircraft manufacturers have included hard-points in the type design, such as nutplates or predrilled holes, for the mounting of harnesses. Some also provide service kits or service instructions that include parts and instructions necessary to install harnesses. If the harness installation does not involve modification of primary structures, it can be returned to service as a minor alteration, unless otherwise specified in the installation instructions. The authorized mechanic needs only to complete a maintenance record entry, and update the equipment list and weight and balance as required.

(2) In some instances, a cable or a bracket can be secured around a structural member, without altering the structure, which will accommodate attachment of the harness. Truss tube construction is most commonly retrofitted with harnesses using this method. After performing static load tests or obtaining stress analysis documentation, the record entries as described above are completed. Refer to chapter 1 to determine design loads.

b. Major alterations can be accomplished by one or more of the following methods and will require completion of the FAA Form 337 using approved data.

(1) STC. There are many STCs issued for installation of shoulder harnesses in a variety of aircraft. The STCs are issued for specific makes and models of aircraft. A listing of STCs can be searched for applicability on the FAA web site.

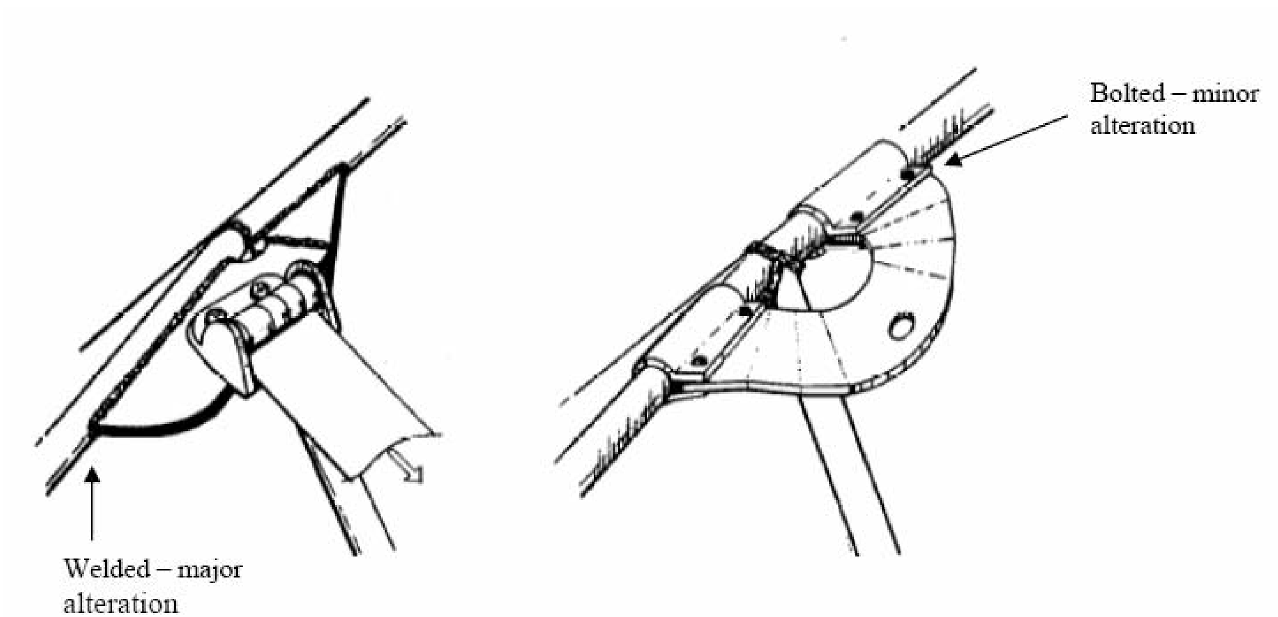
(2) Field approval. An FAA airworthiness aviation safety inspector (ASI) can field approve data under certain conditions. ASIs are not engineers, so unless the case for field approval is supported with adequate data to evaluate the installation, the request for field approval may be turned-down. An example of justification for field approval might be, if an STC exists for the same make/model, with a similar mounting configuration, but the STC does not cover the aircraft by year of manufacture or serial number.

(3) Designated engineering representatives (DER) data. DER can evaluate structural attachments and provide approved data on an FAA Form 8110-3. This data can be substituted for static tests, allowing the installation to be approved for return to service on the FAA Form 337 by an appropriately authorized person.

(4) Manufacturer data. Some manufacturers have developed service kits or service instructions found acceptable or approved by the FAA for installation that involve major alterations to the aircraft. These installations can be approved for return to service on the FAA Form 337 by referencing the kit or instruction number that was installed with no additional approval needed.

(5) Other FAA-approved data. As specified in the Introduction, the data contained herein may conditionally be used as approved data. Data such as static tests, attachments, and materials could be used to show that the installation complies with regulations. Previously performed static tests meeting the minimum standards specified in this AC may be applicable for a retrofit installation if adequately documented.

FIGURE 9-1. MAJOR VS. MINOR ALTERATIONS



904. RESTRAINT SYSTEM CONFIGURATIONS. Restraint systems incorporating a shoulder harness are available in three configurations:

a. The 3-point system consists of a single shoulder belt that is positioned diagonally across the occupant's upper torso. (See Figure 9-2.)

b. The 4-point system consists of dual shoulder belts with one belt passing over each of the occupant's shoulders. Upper end attachment is accomplished at either two locations aft of the occupant's shoulders or at one location aft of the occupant's head after the belts have been joined together in a "Y." Lower end attachment is accomplished at a buckle centered on the occupant's lap belt or symmetrically at each side of the occupant. (See Figure 9-8.)

c. The 5-point system is similar to the 4-point system except that an additional belt, the negative-G strap, commonly referred to as a "crotch strap", is passed between the occupant's legs, attaching one end at the lap belt, and the other end at the front edge of the seat or to the airframe under the seat. A variation of the 5-point system is the 6-point system where 2 belts pass between the occupant's legs. (See Figure 9-9.)

905. ADVANTAGES OF DIFFERENT CONFIGURATIONS.

a. The 3-point single diagonal shoulder harness in combination with a lap belt is the least-cost, most-simple restraint system, and has been proven to work effectively for longitudinal (forward) decelerations. However, during lateral (sideward) decelerations, an occupant in this type of harness has a tendency to slip out and away from the harness even when it fits snugly.

b. The 4-point dual shoulder harness works well for both longitudinal and lateral decelerations.

c. The 5-point system, incorporating the negative-G strap, resists the upward motion of the buckle during loading and limits submarining, which is the tendency for the occupant to slide underneath the lap belt during rapid decelerations. The 5-point system has proven to be very effective and has been adopted for many commercial, agricultural, military, and aerobatic operations.

906. MANUFACTURING STANDARDS. There are several standards that aircraft restraint system may be manufactured to.

a. Restraint Systems Produced by or for the Aircraft Manufacturer. These original equipment manufacturer (OEM) harnesses will bear the manufacturer's part number or other identification.

b. Restraint systems produced under a Parts Manufacturing Approval (PMA) for specific aircraft applications. Restraints will bear the PMA markings required by § 45.15(a).

c. Technical Standard Order (TSO) Restraints. Two TSOs are applicable to the materials and testing of restraint systems. These TSOs do not address attachment to the airframe. TSO restraints will be marked in accordance with § 21.607(d). Aerospace Standard AS8043, Restraint Systems for Civil Aircraft, specifies the test procedures and minimum requirements for the manufacture of civil aircraft restraint systems that conform to these TSOs.

(1) TSO-C22G, Safety Belts, as revised, prescribes the minimum performance standards that safety belts must meet in order to be identified with the applicable TSO marking. This TSO applies to the pelvic or lap belt portion of a restraint system. Belts bearing an earlier revision TSO marking (e.g., C22f) are acceptable if their condition remains satisfactory.

(2) TSO-C114, Torso Restraint Systems, prescribes the minimum performance standards that torso restraint systems must meet in order to be identified with the applicable TSO marking. This TSO applies to pelvic and upper torso restraints and includes the fifth belt of a 5-point system. Harnesses manufactured prior to March 27, 1987, the effective date of this TSO, will not be marked as meeting this TSO.

d. Restraint systems produced under a Military Specification (MIL-SPEC) such as MIL-R-81729. These restraints are marked in accordance with MIL-STD-130.

907. COMPLIANCE WITH STANDARDS.

a. Prior to March 27, 1987, the TSO-C114 shoulder harness standards had not been established.

Lacking such standards, harnesses manufactured prior to this date often were not identified with any markings. These harnesses are acceptable for existing installations if they were installed before the effective date of this TSO and remain in satisfactory condition.

b. For harness installations performed under this AC, use only those restraints that are properly marked and traceable to one of the above standards. There are several restraint manufacturers who can custom-build restraint system components to meet the requirements of TSO-C114. There are also several companies that can manufacture restraints under a PMA for specific applications.

c. It is an acceptable practice to replace existing restraints, including lap belts, with OEM, TSO, or PMA units after compatibility has been determined. This can be performed as preventive maintenance, defined under part 43, Appendix A, paragraph (c)(14), by the aircraft owner/operator, along with the required maintenance record entry. However, if the aircraft is operated under part 121, 129, or 135, the work must be accomplished by appropriately rated mechanic.

d. Used TSO restraints, typically obtained through salvage companies, must be overhauled by an FAA approved facility prior to installation, as these harnesses may have been exposed to unknown environmental conditions or accident loads.

908. MATERIALS. For a more thorough discussion of restraint assembly materials refer to AC 21-34, Shoulder Harness – Safety Belt Installations, as amended.

a. Webbing, the woven fabric portion of the restraint, is made from synthetic materials such as nylon, polyester or Dacron. Minimum breaking strength is determined by the standard under which the restraint was manufactured.

b. Attachment hardware must conform to AN, MS, NAS, or other acceptable industry standards or specifications, and be able to withstand the loads it will be subjected to.

c. Retractors are frequently incorporated into the shoulder harnesses, and sometimes into pelvic restraints. Some current production aircraft incorporate retractors into the harness and the lap belt. Retractors function to provide for adjustment in length and allow the occupant additional freedom of movement when compared to fixed harnesses. They are available in two categories:

(1) Automatic locking retractors provide automatic retraction of webbing for length adjustment and stowage of webbing. Their mechanism permits free webbing extension for coupling of the belt, but the moment any webbing is automatically retracted, the locking mechanism locks to prevent further webbing extension.

(2) Emergency locking retractors are frequently called “inertia reels” because their mechanism provides positive restraint only when inertial forces are experienced. The most common type of inertia reel appropriate for aircraft use is known as the webbing sensitive reel. It produces locking by a change in the rate (acceleration) of webbing withdrawal from the retractor, which is functional for occupant accelerations in any direction producing extension of the webbing. Emergency locking retractors may be equipped with a mechanism that will allow the user to manually lock the reel when a deceleration is anticipated. This feature relieves the possibility of a malfunctioning inertia mechanism.

d. Buckles suitable for shoulder harness installations are defined by their release mechanism and come in three types.

(1) Lift lever

(2) Push Button

(3) Rotary

909. PARTS PRODUCED BY OWNERS/ OTHERS. Section 21.303(b)(2) allows owners or operators to produce parts for altering their own aircraft under certain conditions. Since approved restraints meeting TSO, PMA, or other standards are readily available, applying this regulation for the production of restraints is not advised, as owner produced restraints would also need to demonstrate proof of compliance with these standards. However, airframe attachment brackets or fittings might be candidates to be produced under this regulation. To qualify, the owner must have participated in controlling the design, manufacture, or quality of the part such as:

a. Provide the manufacturer with the design or performance data from which to make the part. The owner could provide the manufacturer with a part to be duplicated.

b. Provide the manufacturer with the materials from which to make the part.

c. Provide the manufacturer with fabrication processes or assembly methods to make the part.

d. Provide the quality control procedures to make the part.

e. Personally supervise the manufacture of the part.

(1) An example of this might be if the owner has discovered a desirable harness attachment bracket in an aircraft similar to his, and he creates a drawing to duplicate this bracket, specifying materials and performance standards. He could then either make the part himself or contract out for the manufacture of the part. When a mechanic installs the part, the maintenance record entry would include the installation information as required from the mechanic, and an entry by the owner that the part was produced under § 21.303(b)(2) by the owner for his aircraft. Note that this regulation provides no authority for the owner to install the part. Furthermore, the mechanic will need to show that the installation meets minimum strength standards through static testing or stress analysis.

(2) Parts and kits to install harnesses are sometimes available from parts warehouses or individuals who supply components and instructions for harness installations without FAA approval such as STC, TSO, or PMA. The installer of these parts will need to obtain FAA approval, typically a field approval, for such installations.

910. THRU 912. RESERVED

SECTION 2. GEOMETRY AND ATTACHMENT

913. GENERAL.

a. Harness attachment points physically locate the shoulder harness relative to the occupant being restrained and establish the angles that will impose loads upon the aircraft structure. Careful selection of appropriate attachment points will maximize overall performance of the restraint system.

b. For best results, the restraint system should be anchored to the primary aircraft structure, defined as: "that structure which contributes significantly to resisting or transmitting flight or ground loads or may lead to an unsafe condition if failed."[1] The structural attachment should be designed to spread the suddenly applied impact loads over as large an area of the structure as possible. The shoulder harness may be attached to selected secondary members that will deform slowly or collapse at a limited rate. This will assist in dissipating the high impact "G" loads to a level tolerable to the human body. However, the possibility of secondary members collapsing and creating a new impact hazard for the occupant, or making it difficult for an occupant to egress from the aircraft should not be overlooked.

914. MOUNTING CONFIGURATIONS. The type of shoulder restraint configuration acceptable for installation is dependent upon the attachments available in the aircraft. Basic harness mounting configurations are:

a. Seat mounted.

b. Airframe mounted.

(1) Side.

(2) Ceiling.

(3) Floor.

(4) Directly rearward.

[1] AC 23-15A, Small Airplane Certification Compliance Program, p. 1.

915. BELT LENGTH. In all installations, it is desirable to keep the harness belt lengths as short as practical while still allowing for the required freedom of movement. Belt stretching, which reduces the effectiveness of the restraint by allowing slack during loading, increases as the belt length does. If too much slack is present, the occupant may contact the instrument panel or slide out of the harness during rapid decelerations.

916. ATTACHMENT GEOMETRY. The following are general guidelines for attachment of 3-point and 4/5-point harness configurations:

a. Single Diagonal 3-Point Harness. A proper installation for this type of harness positions the shoulder belt so that it passes over the midpoint of the shoulder, with the lower end fastened well to the opposite side of the occupant's hip as shown in Figure 9-2. The optimum rearward attachment area for this type of harness is within an angle of 30-degrees above the horizontal measured from the midpoint on the occupant's shoulder as shown in Figure 9-3. Upper harness attachments should be located to the rear and outboard of the occupant's neck. This mounting area is shown in Figure 9-4.

(1) Attachment points inboard of this area would allow the harness to impinge on the neck and could result in neck injury during crash impact. (See Figure 9-5, Neck Impingement.) In addition, the constant rubbing of the strap on the neck would be uncomfortable and, as a result, act as a distraction to the safe operation of the aircraft and a deterrent to use of the harness.

(2) Attachment points forward of this area would reduce the effectiveness of the harness, due to a reduction of contact between the harness and the upper torso of the occupant. (See Figure 9-6, Inadequate Contact.) As a result, the potential for increased forward movement of the torso, increases the likelihood of head impact injuries from the instrument panel. Additionally, the chances for twisting out of the harness are significantly increased.

(3) A retractor can be used with the diagonal harness. It has the added benefit of allowing for more freedom of movement, especially when controls such as fuel selectors are located on the cockpit floor. However, it may be more complicated to mount a retractor and maintain an unrestricted, straight-line entry and exit of the webbing.

b. Double Over-the-Shoulder 4/5-Point Harness. This type of harness should be mounted either directly rearward or to the ceiling. Ideally, the mount should be within the 30-degree vertical angle, or because of the limited number of rearward shoulder harness attachment points in many aircraft, a 5-degree angle below the horizontal is also considered satisfactory, as shown in Figure 9-7. These mounting angles may be used for either the dual independent or the "Y" type belts. Figures 9-8 and 9-9 depict correct harness positioning of 4/5-point harnesses.

(1) If the harness attachment is located more than 5-degrees below the horizontal angle measured from the midpoint on the occupant's shoulder, there is an increased risk of spinal compression caused by the vertical component induced during impact deceleration. (See Figure 9-6, Spinal Compression.)

(2) For dual independent harnesses, the outboard limit must be established to provide sufficient contact that will prevent the belt from slipping off the shoulder, and the maximum inboard angle is limited to a point that will prevent impingement on the neck. (See Figure 9-5.) Where the mounting structure is incapable of withstanding loads imposed by a "Y" type harness single retractor, two retractors used with dual independent harnesses may spread the load enough to satisfy strength requirements. Dual independent harnesses that are crossed aft of the occupant's head will need careful evaluation to preclude neck impingement.

(3) Retractors are often used with "Y" type and dual independent harnesses. The single retractor used with a "Y" type harness is mounted aft of the center vertical plane of the occupant. (See Figures 9-4 and 9-10.) The fifth belt, if used, is attached to the seat or airframe so that it joins the buckle perpendicular to the lap belt in the centerline of the seat. Figure 9-9 shows a 5-point system.

FIGURE 9-2. 3-POINT SINGLE HARNESS POSITIONING

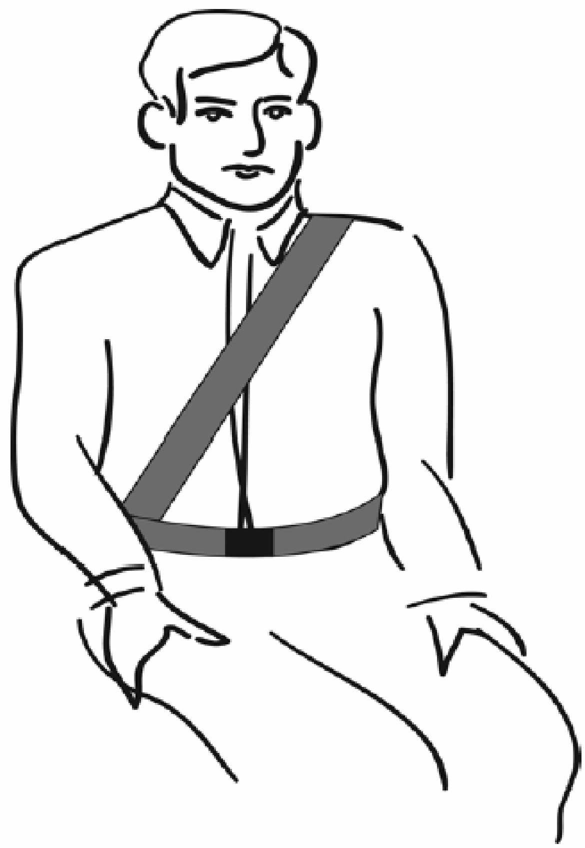


FIGURE 9-3. SIDE MOUNTED SINGLE DIAGONAL TYPE HARNESS

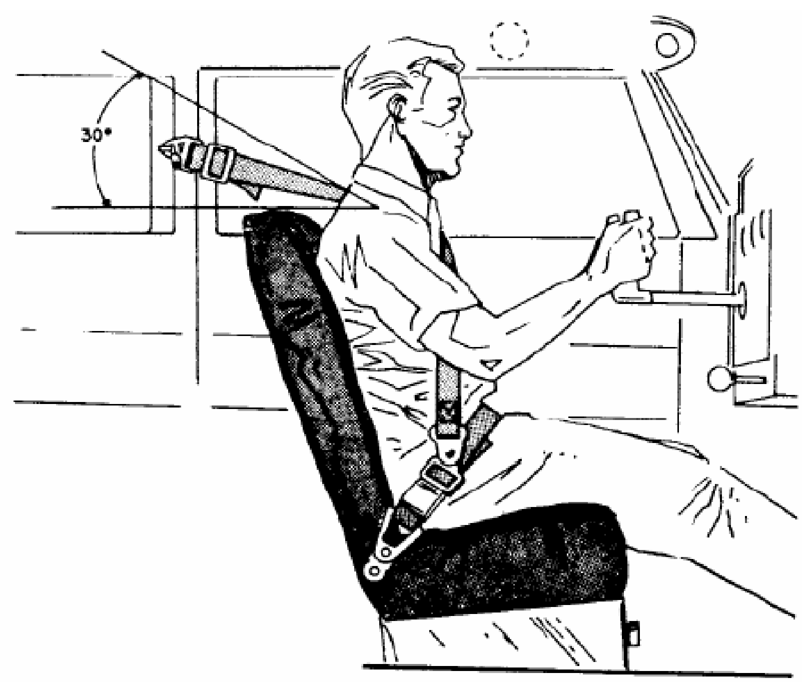


FIGURE 9-4. ACCEPTABLE HARNESS MOUNTING AREAS

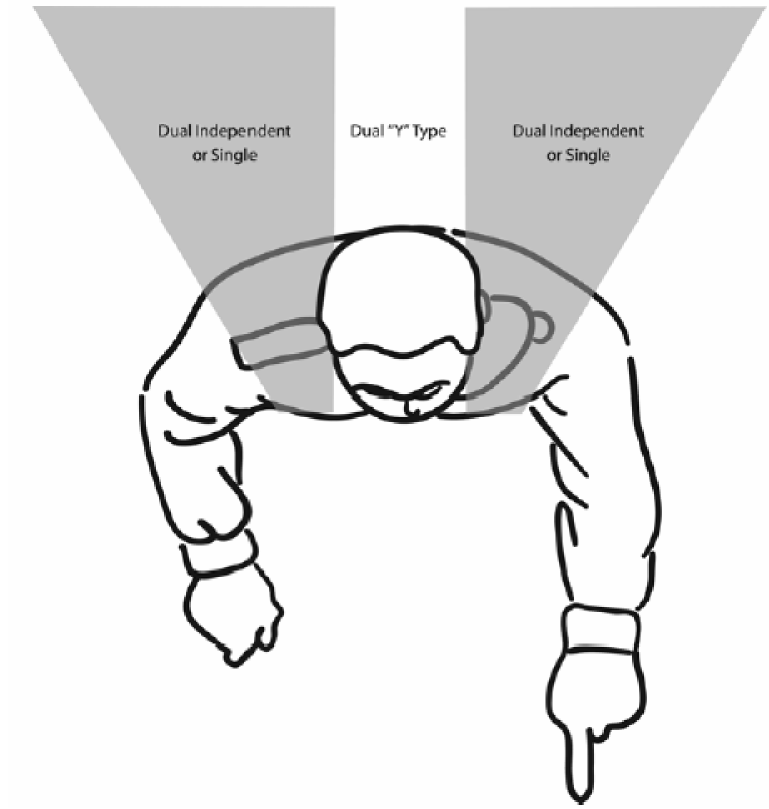


FIGURE 9-5. UNDESIRABLE HORIZONTAL HARNESS POSITIONING

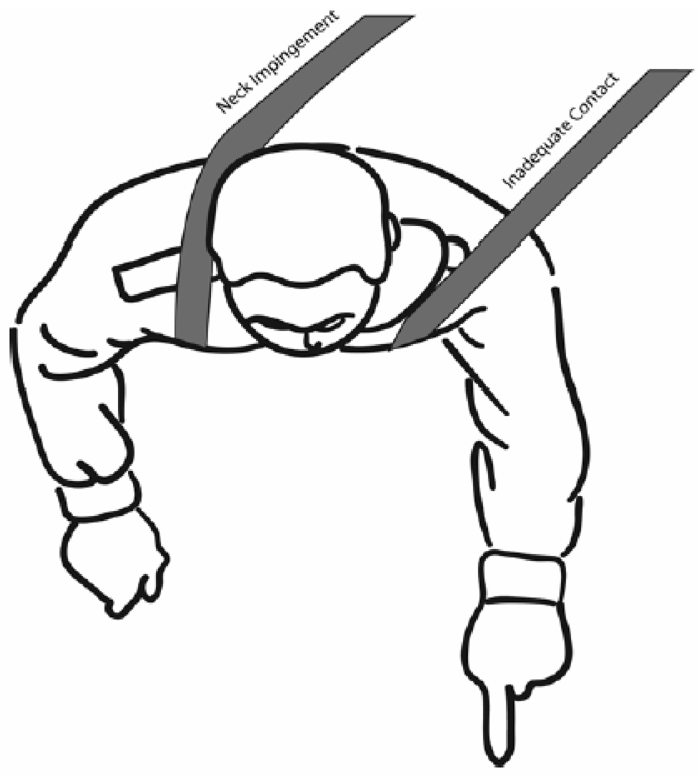


FIGURE 9-6. UNDESIRABLE VERTICAL HARNESS POSITIONING

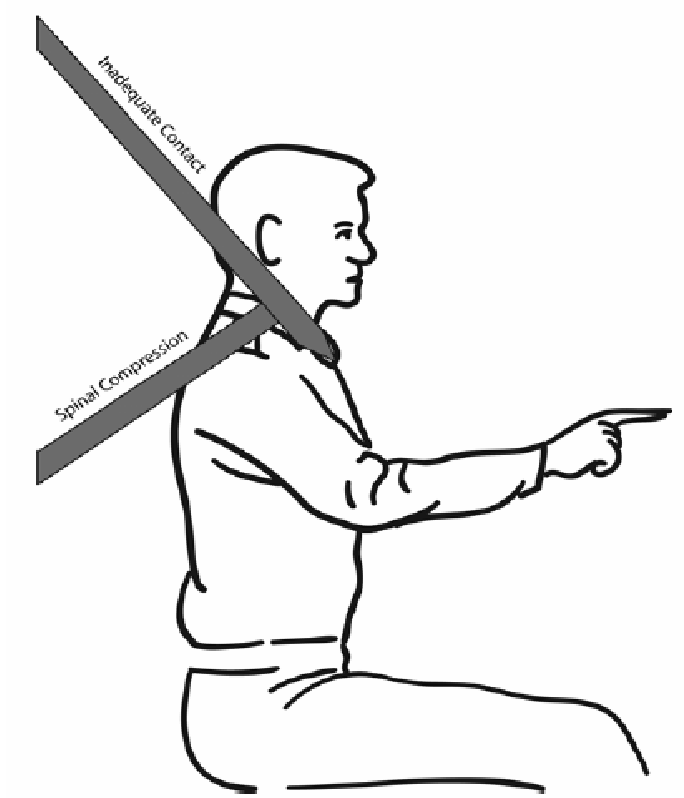


FIGURE 9-7. CEILING MOUNTED INERTIA REEL – DOUBLE OVER-THE-SHOULDER TYPE HARNESS

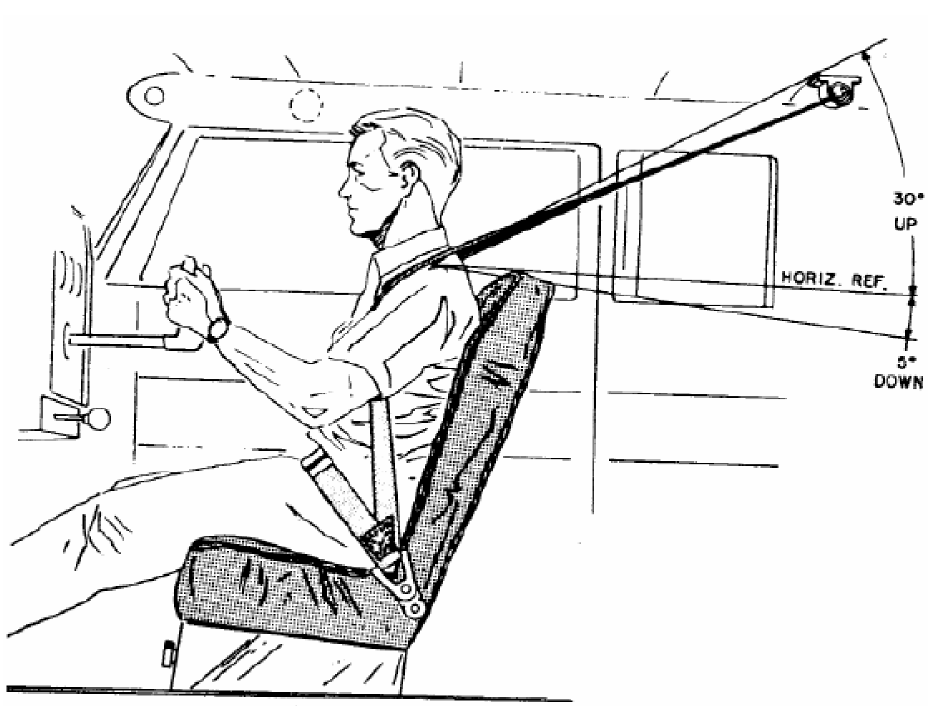


FIGURE 9-8. 4-POINT INDEPENDENT HARNESSES POSITIONING

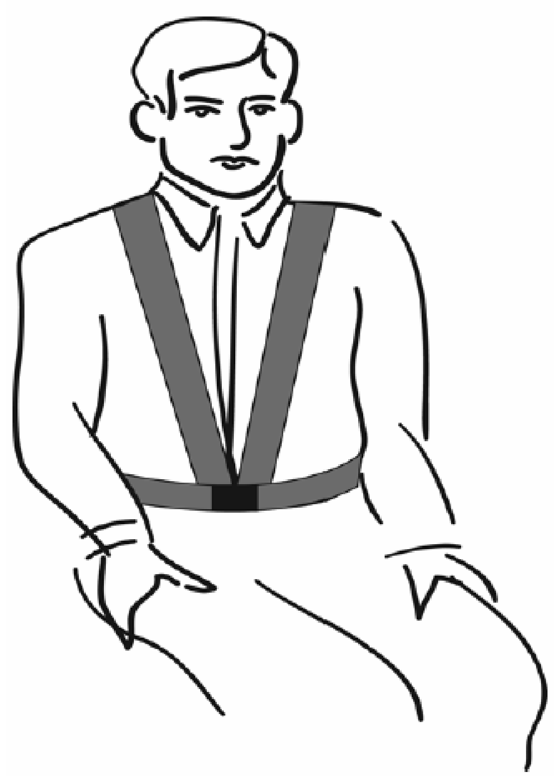


FIGURE 9-9. 5-POINT HARNESS POSITIONING

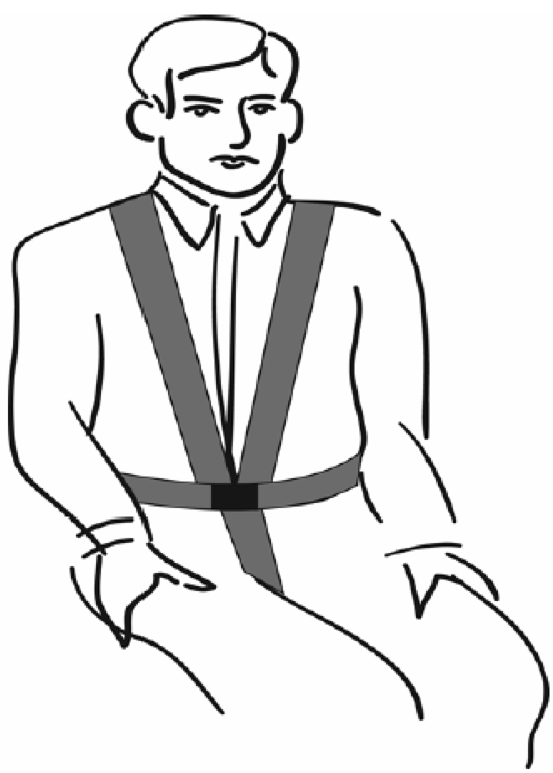
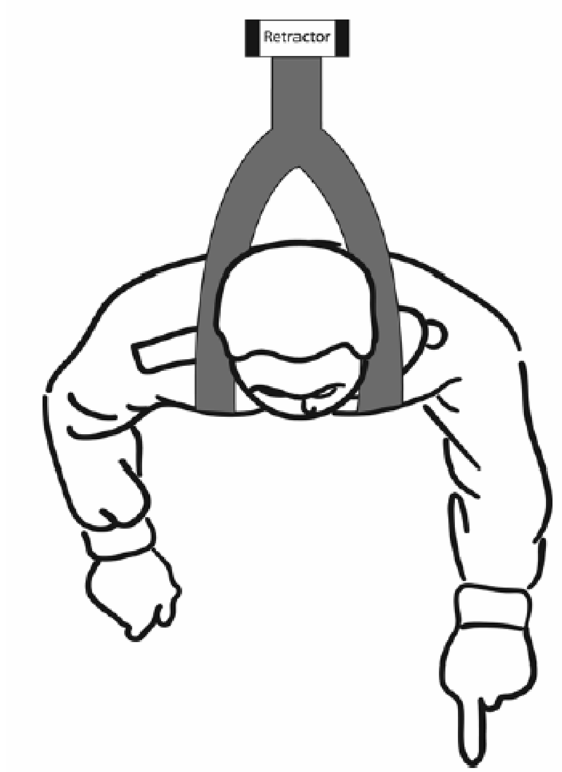


FIGURE 9-10. "Y" TYPE RETRACTOR POSITIONING



917. AREA AND ANGLE DEVIATIONS. While the areas and angles given in the above paragraphs are intended to assist in the selection of attachment points, they should be considered the desirable optimum. Area and/or angle deviations could result in a decrease in the overall efficiency of the restraint system; however, they may be necessary in order to permit harness installation in an aircraft that cannot accommodate ideal harness geometry. As discussed in section 3, angles have a significant bearing on static test load requirements.

918. STRUCTURAL ATTACHMENT. This chapter presents only a few generic examples of structural attachment of harnesses that may be determined to be acceptable for a specific application. Refer to section 3 for strength requirements of the attachment.

a. Floor and Seat Attachments. The dual over-the-shoulder or Y type harness may be used with either floor or seat mounting points. Typical installations are illustrated in Figures 9-11, 9-12, and 9-13. Several factors need to be considered that may make this configuration undesirable:

(1) The floor, seat structure, and anchorages must be capable of withstanding the additional "G" loads imposed upon them by the restraints.

(2) The height of the seat back should at least be equal to the shoulder height of the seated occupant to reduce the possibility of spinal compression injuries.

(3) Harness guides may be necessary to maintain proper harness position over the occupant's shoulders.

(4) Seat back strength is critical to performance of these installations. Folding seat backs must have a locking mechanism that can withstand the imposed loads; however, a locked seat back may impede emergency egress of other occupants and create an impact hazard for occupants seated aft of the locked seat back.

FIGURE 9-11. FLOOR MOUNTED INERTIA REEL – DOUBLE OVER-THE-SHOULDER TYPE HARNESS

FIGURE 9-12. TYPICAL FLOOR MOUNTED INERTIA REEL INSTALLATION

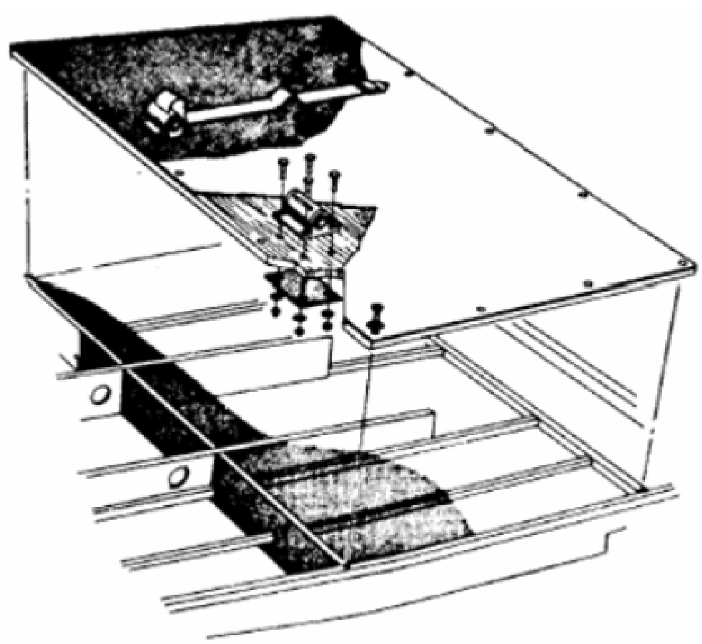
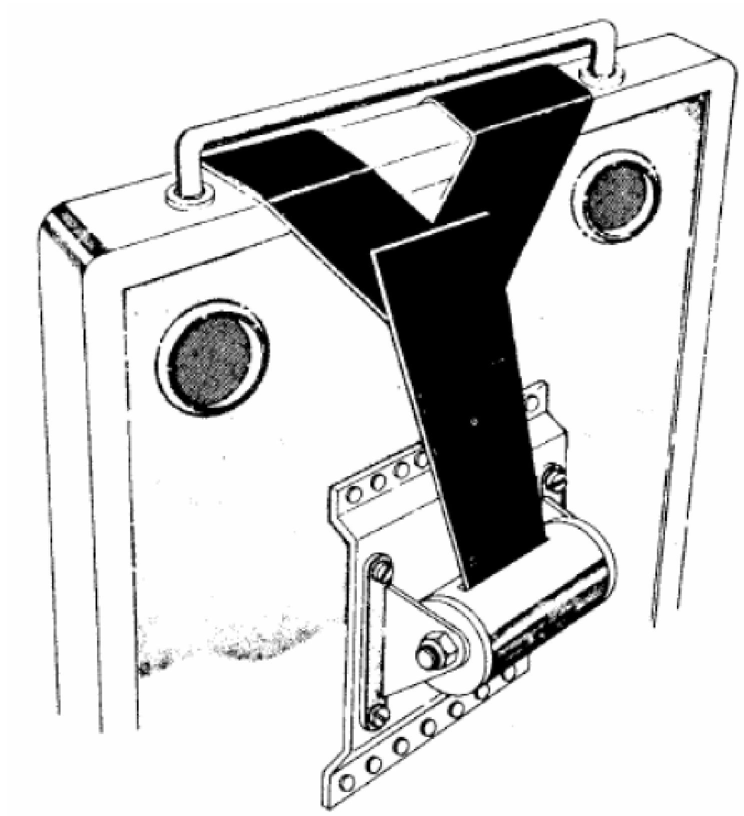


FIGURE 9-13. TYPICAL SEAT MOUNTED INERTIA REEL INSTALLATION



b. Airframe Attachments. The method used for the attachment of shoulder harness anchorages is dependent upon the construction features of the aircraft involved. When selecting a point of attachment, careful consideration should be given to the static strength and testing requirements found in section 3.

(1) Monocoque/semimonocoque type constructions. Illustrations of typical aircraft members and installation methods are shown in Figures 9-14 through 9-18. Intercostal doublers and stiffeners are frequently added to provide increased strength. AC 43.13-1, Acceptable Methods, Techniques, and Practices-Aircraft Inspection and Repairs, chapter 4 (as amended), provides useful information on fabrication and installation of sheet metal repairs that may be applicable to doubler and stiffiner installations.

(2) Tube type construction. Various typical methods of attaching shoulder harness anchorages are shown in Figure 9-19. Attachment should be accomplished at the intersection of tubular members and not at the center of single unsupported tubes. When aircraft cable is used as a component in a shoulder harness anchorage, swage the cable terminals in accordance with the procedures contained in AC 43.13-1, chapter 7, as amended.

c. Structural Repair Instructions. In some instances, structural repair instructions are provided in the aircraft manufacturer's maintenance manual. While these instructions are primarily intended for use in repairing defective or damaged structure, they may also be used as reinforcement methods for shoulder harness attach fittings.

d. Flexible Attachments. Various aircraft are designed so that fuselage members and/or skin will flex or "work." This type of structure should not be heavily reinforced for the purpose of attaching shoulder harnesses, as this would defeat the design purpose. In these cases, use a localized reinforcement such as that shown in Figure 9-18, at the attachment point. This will allow the fuselage to flex while still maintaining a collapsible structure to absorb the loads encountered in a crash.

919. THRU 922. RESERVED

FIGURE 9-14. TYPICAL WING CARRY-THROUGH INSTALLATION

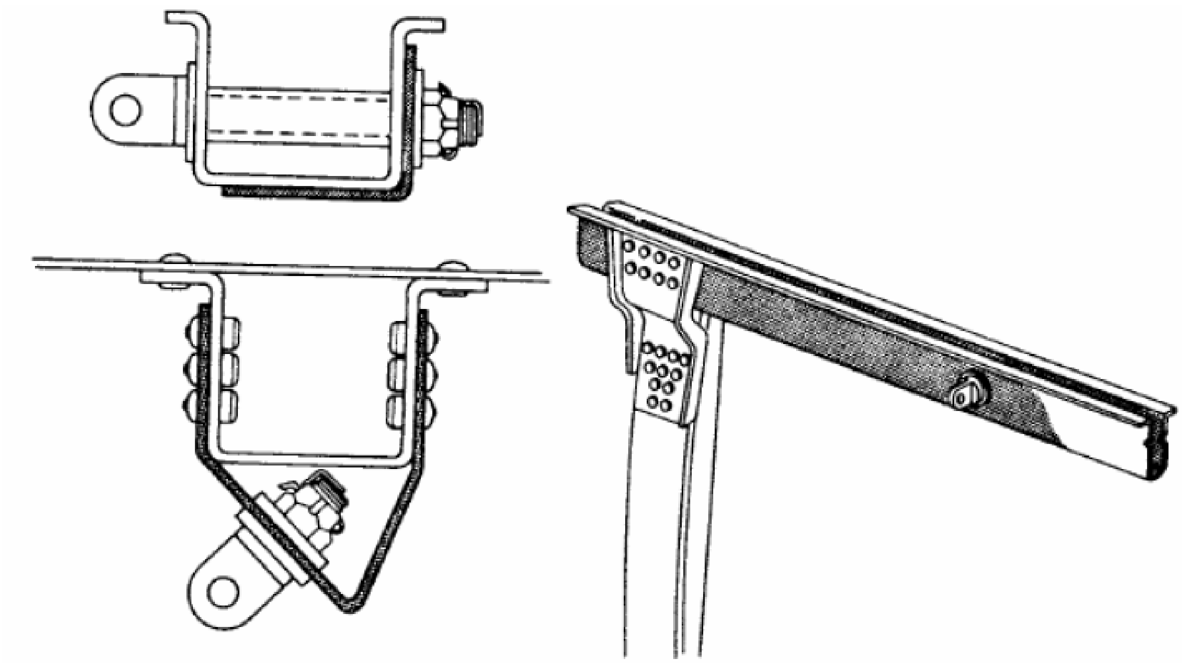


FIGURE 9-15. TYPICAL HAT SECTION REINFORCEMENT INSTALLATION

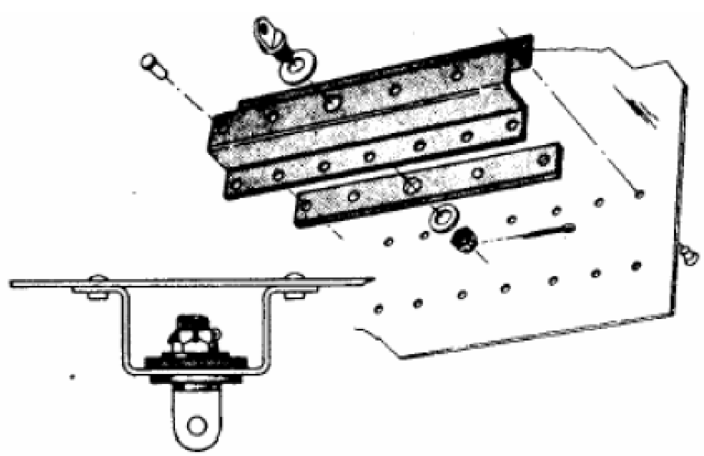


FIGURE 9-16. TYPICAL BULKHEAD REINFORCEMENT INSTALLATION

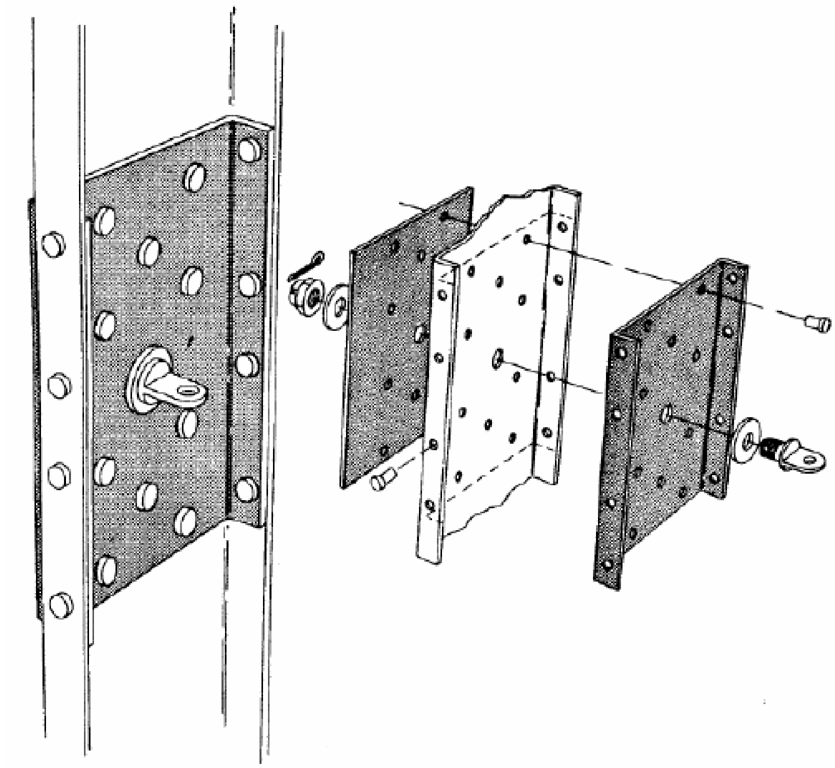


FIGURE 9-17. TYPICAL STRINGER SECTION REINFORCEMENT INSTALLATION

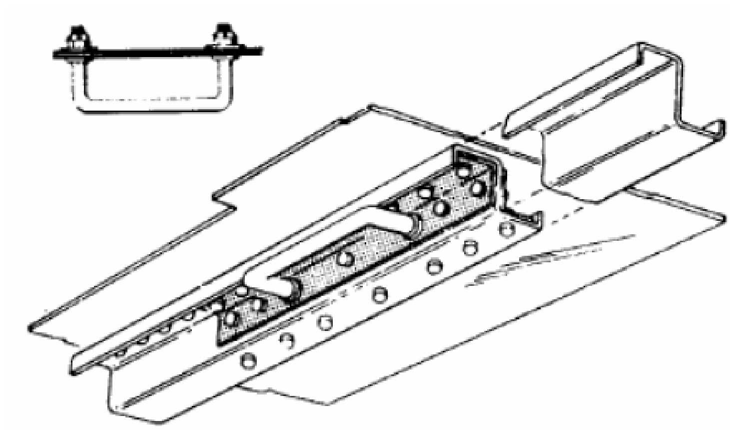


FIGURE 9-18. TYPICAL LOCALIZED STRINGER SECTION REINFORCEMENT INSTALLATION

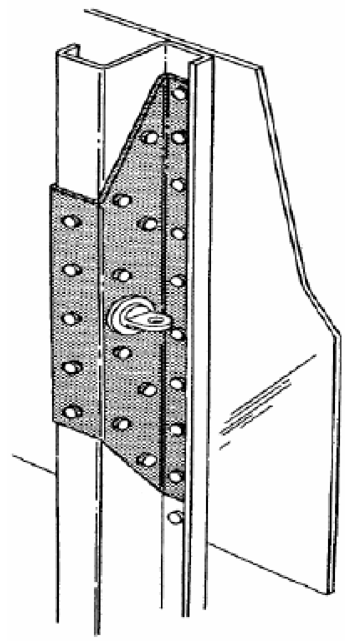
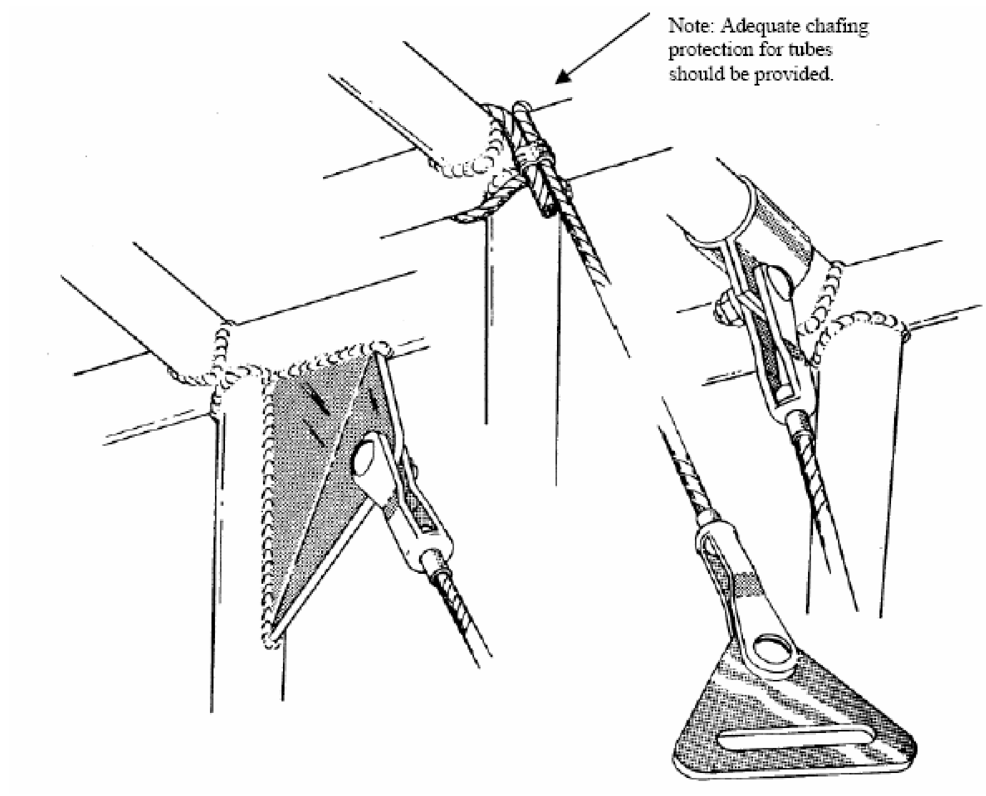


FIGURE 9-19. TYPICAL ATTACHMENTS TO TUBULAR MEMBERS



SECTION 3. STATIC STRENGTH AND TESTING

923. PROOF OF COMPLIANCE. Retrofit harness installations must be proven to meet minimum strength requirements applicable to the aircraft certification basis; otherwise, their ability to provide adequate restraint under accident decelerations is undetermined. Static strength requirements can be demonstrated by static test or stress analysis.

a. Static tests are accomplished by application of the ultimate load to the harness attachment point. Since TSO and PMA restraints have already been shown to comply with strength requirements, static tests are accomplished to substantiate the airframe attachment. Static testing in operational aircraft is undesirable because there is the risk of airframe damage; however, static testing in a conforming fuselage or cabin section of the same make and model aircraft is an acceptable alternative.

b. Stress analysis is accomplished through engineering data developed for the specific attachment point. This data is developed by FAA Designated Engineering Representatives (DER) with aircraft structural approval authority. The DER will perform the analysis, prepare a report, and complete FAA Form 8110-3, Statement of Compliance with the Federal Aviation Regulations, which approves the data for structural attachment of the harness. Since the DER is only approving the structural aspects of harness attachment, approval for return to service for the installation as a whole is accomplished by the authorized individual or agency on FAA Form 337. To locate an appropriately authorized DER in your locality, contact the FAA Flight Standards District Office or Aircraft Certification Office having jurisdiction in your geographic area.

NOTE: It is up to the installer to determine which method will be the most efficient. For simple installations, it is possible that the DER could perform stress analysis in just a few hours. The DER may also be able to assist in the design of attachment points.

924. LOAD AND DISTRIBUTION — AIRPLANES. Table 9-1 provides a reference to determine minimum static test loads based upon certification date, category, standard occupant weight, fitting factors, and harness factors. The numbers in each line are multiplied together to obtain the static test load minimum standard.

a. Load. For airplanes type certificated prior to September 14, 1988, occupant weights were established as 170 pounds for normal category and 190 pounds for utility and acrobatic category airplanes. The additional 20 pounds accounts for a parachute. For aircraft type certificated on or after this date, 215 pounds is the established weight for the 3 categories.

b. Distribution. In the assessment of a combined shoulder harness-safety belt restraint system, a forward static test load distribution of 40 percent to the shoulder harness and 60 percent to the safety belt has been an acceptable combined static test load distribution. Therefore, demonstration of harness attachment static strength is accomplished by applying 40 percent of the ultimate restraint system load to a single or “Y” type harness attachment, or 20 percent of the load simultaneously to each attachment of a dual independent harness. Static strength test load requirements for the fifth belt of a 5-point system have not been established for aircraft.

c. Factor. The airplane must be designed to protect each occupant during emergency landing conditions. The critical ultimate static load is established at 9.0 G’s in the forward direction. If the attachment can sustain this load, it is assumed the lesser loads in upward, sideward, and downward directions will be accommodated.

d. Fitting Factors. Regulations prescribe multiplying factors of safety, fitting factors, that structural attachments must be subjected to in order to prove minimum load bearing capabilities. A 1.15

fitting factor for restraint attachments in small aircraft type certificated prior to September 14, 1969, must be used. For aircraft type certificated on or after this date, a fitting factor of 1.33 must be used.

925. LOAD AND DISTRIBUTION — NORMAL CATEGORY HELICOPTERS.

a. Load. For helicopters type certificated in the normal category, occupant weights are established as 170 pounds.

b. Distribution. In the assessment of a combined shoulder harness-safety belt restraint system, for helicopters type certificated prior to December 13, 1989, a forward static test load distribution of 60 percent to the shoulder harness and 60 percent to the safety belt is the required combined static test load distribution.

c. G Factor. For helicopters type certificated prior to December 13, 1989, the critical ultimate static load is established at 4.0 G's in the forward direction. If the attachment can sustain this load, it is assumed the loads in upward, sideward, and downward directions will be accommodated.

d. Fitting Factors. A 1.15 fitting factor for restraint attachments in helicopters type certificated prior to December 16, 1984, must be used. For helicopters type certificated on or after this date, a fitting factor of 1.33 must be used.

NOTE: Due to increased G factors and limited applicability for retrofit harnesses installations, helicopters type certificated on or after December 13, 1989, are not addressed in this chapter. Additionally, airplanes certificated on or after September 14, 1988, must undergo dynamic testing which is not covered in this chapter.

926. DEMONSTRATING STATIC LOADS. Two methods are available to demonstrate compliance with static load requirements. One method involves the fabrication of test blocks and application of the ultimate belt load to the blocks that are being restrained by the harness. The other method involves use of an equation to calculate and simulate the ultimate belt load at the harness attachment fitting, compensating for horizontal and vertical angles as a result of restraint attachment geometry.

a. Test Blocks. Although AC 23-4, Static Strength Substantiation of Attachment Points for Occupant Restraint System Installations, was cancelled on January 27, 2003, it is still available in various archives and can be used as an acceptable method for fabricating test blocks and applying static loads to restraints. Using this method, deceleration loads imposed upon the restraint by the occupant are accurately duplicated, and the load figures in Table 9-1 can be applied directly to the test blocks. Because of the time-consuming nature of fabricating test blocks for this procedure, it may be desirable to use the second method described below that, in effect, replaces the test blocks with an equation.

b. Calculation. The alternate method is to perform a mathematical calculation to account for the angles that the attachment points are subjected to during decelerations. Increases in angles can result in a significant increase in the load that must be applied to simulate belt loading during occupant decelerations. It is not uncommon for a harness to impose both vertical and horizontal angles that must be compensated for by use of the equation. (See Figures 9-20 and 9-21.) In the case of a "Y" type harness, the horizontal angle is eliminated as long as the attachment is located along the vertical centerline of the occupant.

(1) The equation used to perform the calculation is:

Test Load = [Static Test Minimum]
[1/Cosine(vertical angle)]
[1/Cosine(horizontal angle)]

(2) Table 9-2 provides 1/Cosine numbers for various angles. These numbers can be inserted into the equation to determine the static load that needs to be applied.

For example, using Table 9-1 and Table 9-2, if the airplane is normal category, post September 14, 1969, pre September 14, 1988, the test load is 814 lbs. If the horizontal angle is 10^{o} and the vertical angle is 15^{o}, the Belt load is calculated as follows.

Test Load = [814][1/Cos(15)][1/Cos(10)]
= [814][1.035][1.015]
= 855 lbs.

This is not a large increase, but if both angles were 30^{o}, the belt load would be 1086 lbs.
[814][1.155][1.155] = 1086

If there is no angle present in one of the directions, that portion of the equation is eliminated. For an airplane type certificated in the normal category prior to September 14, 1969, with a vertical angle of 25 degrees, the equation would be: [704][1.103] = 777 lbs.

c. After obtaining the test load value, the load is applied for a minimum of 3 seconds, forward (horizontally) when using test blocks, or in the direction of the restraint angles when applying the load to the attachment point(s).

d. Equipment used for measuring test loads must be calibrated to a standard acceptable to the FAA. Strain gauges are capable of performing these measurements; however, other equivalent equipment as determined by the user may be acceptable.

927. THRU 930. RESERVED

TABLE 9-1. STATIC TEST REQUIREMENTS

TC Basis Airplanes	Category*	G Factor	Weight Factor (lbs)	Fitting Factor	Harness Factor (40%)	Static Test Minimum Load (lbs)
Effective 09/14/88 and on	N, U, A	9	215	1.33	.4	1029
Effective 09/14/69	N	9	170	1.33	.4	814
	U, A	9	190	1.33	.4	910
Pre 09/14/69	N	9	170	1.15	.4	704
	U, A	9	190	1.15	.4	787
Helicopter Pre 12/16/84	N	4	170	1.15	.6	469
Helicopter Post 12/16/84 Pre 12/13/89	N	4	170	1.33	.6	543

* N=Normal, U=Utility, A=Acrobatic

TABLE 9-2. CALCULATED 1/COSINE VALUES

Degrees	5	10	15	20	25	30	35	40	50	60
1/Cos Multiplier	1.004	1.015	1.035	1.064	1.103	1.155	1.221	1.305	1.556	2.0

FIGURE 9-20. VERTICAL ANGLE

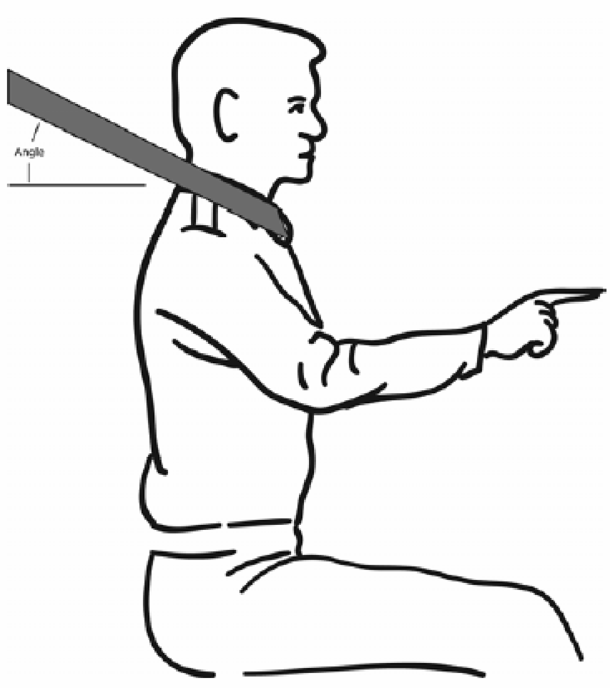
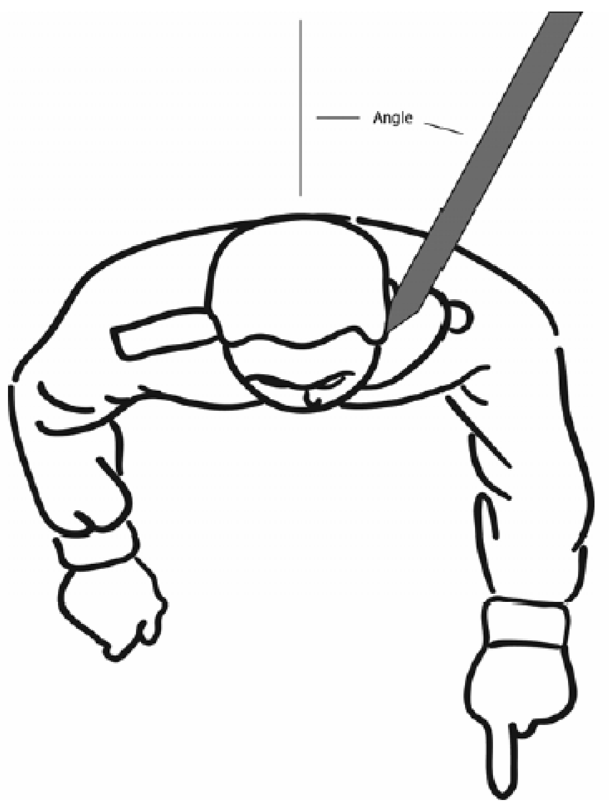


FIGURE 9-21. HORIZONTAL ANGLE



SECTION 4. INSTALLATION AND INSPECTION CHECKLISTS

931. GENERAL. The following checklists are not all-inclusive, but may provide a quick reference for factors to consider when installing retrofit shoulder harnesses and for establishing instructions for continued airworthiness of retrofit harness installations. If available, manufacturer's instructions take precedent over this AC.

a. Shoulder Harness Installation Checklist.

(1) Locate potential attach points.

(2) Evaluate the geometry of attach points in relation to the occupant and determine the best harness configuration. Short belt lengths are desirable.

(3) Ensure that the configuration will allow unimpeded egress for all occupants. Single-point release is mandatory. There should be a means to secure belts and harnesses when not in use to avoid interfering with operation of the airplane or egress from it.

(4) Restraints must allow crewmembers to perform all necessary flight operation functions while seat belts and harnesses are fastened.

(5) Utility and acrobatic category airplane restraints must accommodate an occupant wearing a parachute.

(6) If retractors will be used, ensure that mounting will provide straight-line entry and exit of webbing to prevent binding or frictional drag. Retractor loading should not impose bending forces upon retractor mounting brackets.

(7) If harness installation will impose loads on the seat back or seat attachment, additional evaluation will be needed for structural integrity.

(8) Refer to the aircraft type certificate to find the certification date and basis to establish minimum static load requirements. Determine the feasibility of performing static tests or stress analysis. Static testing on operational aircraft is not recommended.

(9) Select only restraints that meet OEM, TSO, PMA, or other established industry standards or specifications.

(10) Install restraints using AN, MS, NAS, or other acceptable aircraft hardware with adequate strength properties.

(11) Perform a check to ensure all components function properly.

(12) Complete record entries as required, including FAA Form 337, logbook, weight and balance, and equipment list.

b. Shoulder Harness Inspection Checklist. Inspect restraint system for condition and function at each annual or each required inspection of cabin or cockpit equipment. If questions arise regarding any of the conditions listed below, contact the restraint manufacturer for specific limits.

(1) Inspect stitching on webbing for broken or missing stitches.

(2) Inspect webbing for fraying, fading or cuts. Fraying that causes binding in the retractor is excessive. Fading caused by exposure to sunlight or chemicals may indicate a reduction in strength.

(3) Ensure that TSO, PMA or other required identification tags are present and legible on each belt assembly. Tags may not be required for harnesses manufactured prior to March 27, 1987. Contact the aircraft or restraint manufacturer for specific requirements.

(4) If so equipped, check the harness post on the lap belt to make sure the nylon bushing or grommet is present and functional to provide positive harness end fitting engagement. Do not replace the bushing with nylon bundle ties. Contact the belt manufacturer for replacement bushings/grommets and instructions.

(5) Check the belt buckle for latching and release functions.

(6) Check the retractors to ensure that locking mechanisms engage when webbing is positioned at the normal operating length. Inertia reels should lock when the webbing is accelerated quickly out of the reel. Automatic locking retractors should lock at approximately one-inch increments as the webbing retracts back into the reel.

(7) Inspect buckles, connectors, and fittings for corrosion, cracks, or other damage. Check mounting hardware for security and ensure that airframe attachment of end fittings self-align and do not bind.

(8) Inspect quick release end fittings for cotter pins.

(9) If equipped with manual locking inertia reel, check for proper operation and condition.

932. THRU 999. RESERVED.

CHAPTER 10. AIRCRAFT BATTERY INSTALLATIONS

SECTION 1. GENERAL

1000. PURPOSE. This section contains structural and design considerations for the fabrication of aircraft battery installations.

1001. HAZARDS AND WARNINGS: HANDLING PRECAUTIONS. Serious injury can result from carelessness while handling and working with batteries. Failure to heed these warnings could result in serious injury or death.

a. All tools must be insulated.

b. Care must be taken with all metal items to include clothing items such as belt buckles, zippers, metal fasteners and wallet chains, as well as jewelry items such as rings, watches, bracelets, and necklaces. All metal or conductive articles should be removed from your person when handling batteries.

c. Wear protective clothing and eye protection. The electrolyte can cause burns if in contact with skin or eyes. Do not touch eyes, nose, or mouth after handling batteries or acid.

d. Do not smoke or hold naked flames near batteries on charge. If allowed to accumulate in a confined space, the gases emitted during charge could cause an explosion. To prevent the accumulation of hydrogen gas in the manifold, do not charge a flooded electrolyte or vented battery on the bench with the cover on.

e. Do not service flooded or vented lead-acid and nickel-cadmium batteries in the same shop area, as cross contamination of acid and alkaline electrolytes may happen.

f. Always pour acid into water, NEVER pour water into acid.

g. Do not use petroleum spirits, trichloroethylene or other solvents.

h. Know the location and use of emergency eyewash and shower nearest the battery charging area.

1002. ADDITIONAL REFERENCES.

a. AC 20-106, Aircraft Inspection for the General Aviation Aircraft Owner.

b. AC 43-4, Corrosion Control for Aircraft.

c. AC 120-27, Aircraft Weight and Balance Control.

d. FAA-H-8083-1, Aircraft Weight and Balance Handbook

e. FAA-H-8083-19, Plane Sense General Aviation Information.

1003. LOCATION REQUIREMENTS. The battery installation and/or its installation should provide:

a. Accessibility for Battery Maintenance and Removal. The installation should ensure that the battery could be easily installed or removed and serviced without removing cowling, seats, fairings, etc.

b. Protection from Engine Heat. Protection from Engine Heat: If installed in the engine compartment the battery should be protected from extreme heat both during engine operation and after the engine has been shut down. This kind of protection can be provided by a source of cooling air to the battery box or additional thermal protection around the battery. Care should be taken not to interfere with the flow of engine-cooling air.

c. Protection from Mechanical Damage. Vibration and other shock loads are a major cause of short battery life. Install the battery in a location that will minimize damage from airframe vibration and prevent accidental damage by passengers or cargo.

d. Passenger Protection. Insure that the battery is enclosed within a container/box so that passengers/crew are protected from any fumes or electrolytes that may be spilled as a result of battery overheating, minor crash, or un-intentional inverted flight.

e. Airframe Protection. To minimized damage to adjacent metal structures, fabric covering or electrical equipment can be accomplished by properly locating battery drains and vent discharge lines, and adequately venting the battery compartment. To protect the airframe structure and fluid lines apply asphaltic or rubber-base paint to the areas adjacent to and below the battery or battery box.

1004. AIRCRAFT STORAGE BATTERY DESIGN AND INSTALLATION.

a. Lead Acid.

(1) Each TSO approved aircraft storage battery must be designed as required by regulation and installed as prescribed by the manufacturer.

(2) No explosive or toxic gases emitted by any battery in normal operation, or as the result of any probable malfunction in the charging system or battery installation, may accumulate in hazardous quantities within the airplane.

(3) Corrosive fluids or gases that may escape from the battery may damage surrounding structures or adjacent essential equipment.

b. Nickel Cadmium.

(1) Each TSO approved storage battery must be designed as required by regulation and installed as prescribed by the manufacturer.

(2) Safe cell temperatures and pressures must be maintained during any probable charging and discharging condition. No uncontrolled increase in cell temperature may result when the battery is recharged (after previous complete discharge).

(a) At maximum regulated voltage or power;

(b) During a flight of maximum duration; and

(c) Under the most adverse cooling condition likely to occur in service.

(3) Compliance with paragraph (N)(2) must be shown by tests unless experience with similar batteries and installations has shown that maintaining safe cell temperatures and pressures presents no problem.

(4) No explosive or toxic gases emitted by any battery in normal operation, or as the result of any probable malfunction in the charging system or battery installation, may accumulate in hazardous quantities within the airplane.

(5) Corrosive fluids or gases that may escape from the battery may damage surrounding structures or adjacent essential equipment.

1005. DUPLICATION OF THE MAN-UFACTURER'S INSTALLATION. The availability of readymade parts and attachment fittings may make it desirable to consider the location and/or type of installation selected and designed by the airframe manufacturer. Appreciable savings in time and work may be realized if previously approved data and/or parts are used.

1006. OTHER INSTALLATIONS. If the battery installation has not been previously approved, or if the battery is to be installed or relocated in a manner or location other than provided in previously approved data, perform static tests on the completed installation as outlined in chapter 1. Because of the concentrated mass of the battery, the support structure should also be rigid enough to prevent undue vibration or undue shock, which may lead to early structural failure.

1007. DELIVERY INSPECTION. When the battery is unpacked, a thorough inspection should be made to ensure that no damage occurred during shipment. Inspect the shipping container as well as the battery. Before putting the battery into service, perform a safety check by following these points carefully.

a. Damage. See if any liquid has spilled into the shipping container. This may indicate that a cell is damaged. Check for dented, cracked, or discolored areas on the sides and bottom of the battery case. Check for cracked cell cases or covers. Do not place a damaged battery into service.

b. Shorting Straps. Some nickel-cadmium batteries are shipped with shorting devices across the main power receptacle output terminals. Before subjecting a battery to electrical service this device must be removed.

c. Electrical Connections. Test all terminal hardware to ensure tightness. Poor electrical contact between mating surfaces may reduce discharge voltage, cause local overheating and damage the battery.

d. Before charging the battery read and become familiar with the manufacturer's charge procedures.

1008. BATTERY INSTALLATION AND REMOVAL. The following instructions are generic. See the airframe manufacturer's maintenance manuals or STC for instructions specific to a particular aircraft model.

a. Removal.

(1) Set master switch to the OFF position and tag the switch.

(2) Disconnect any external power supply.

(3) Open battery compartment access panels.

(4) Disconnect battery quick disconnect plug or remove terminal bolts and disconnect battery cables from battery terminals. Always disconnect the ground cable first and install the ground cable last.

(5) Disconnect battery ventilation tubes, if any.

(6) Unlock battery hold down clamps or remove battery hold down bars. Disengage battery.

(7) Carefully remove battery. WARNING: Batteries are heavy. Use appropriate lifting devices or equipment. Use battery handles where provided.

b. Installation.

(1) Inspect the battery for damage. Cracks in metal or plastic containers are not permitted. Dents in metal containers that impinge on the interior plastic container are not acceptable.

(2) Set master switch to the OFF position and tag (explain what should be on the tag).

(3) Disconnect any external power supply and tag (explain what should be on the tag).

(4) Open battery compartment access panels.

(5) Ensure the battery container or tray is clean and dry. Treat and paint any corrosion areas.

(6) Install battery in battery container or tray. WARNING: Batteries are heavy. Use appropriate lifting devices or equipment. Use battery handles where provided.

(7) Engage battery hold down hardware, torque and safety wire per airframe manufacturer's maintenance manual.

(8) Connect battery vent tubes.

(9) Connect battery quick disconnect plug, any auxiliary connector or for ring terminals, install with bolt and lock washer. Torque terminal bolts as recommended by the manufacturer.

(10) Replace electrical compartment access panel.

(11) Perform an operational test.

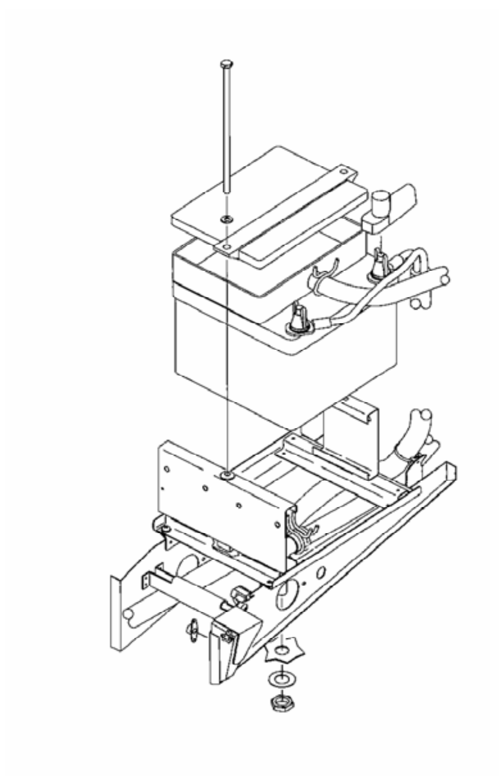
(12) Update aircraft weight and balance data, if necessary.

(13) Update equipment list, if applicable.

(14) Make a log book entry with battery serial number and date of installation.

1009. THRU 1012. RESERVED.

FIGURE 10-1. TYPICAL BATTERY INSTALLATION IN THE AIRCRAFT



SECTION 2. LEAD ACID BATTERY INSTALLATIONS

1013. GENERAL. In a lead acid battery the voltage will slowly drop as opposed to a nickel-cadmium battery. These batteries are typically less expensive, do not require temperature sensor monitoring, require virtually no maintenance, and when unable to meet the manufacturer's capacity requirements, are simply removed and replaced.

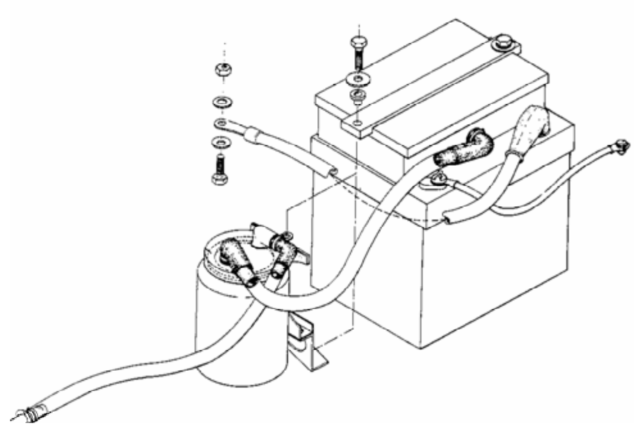
1014. BATTERY BOX. The battery box should have an open drain in case the electrolyte overflows. The box should be vented to prevent an accumulation of flammable hydrogen gas.

1015. SECURING THE BATTERY. Install the battery in such a manner as to hold the battery securely in place without subjecting it to excessive localized pressure, which may distort or crack the battery case. Apply paralketone, heavy grease, or other comparable protective coating to battery cables terminal nuts/connectors. Ensure that proper torque is applied to the terminal nuts/connectors. Do not over tighten terminal nuts, which may result in fracturing of the terminal posts. Provide adequate clearance between the battery and any bolts and/or rivets which may protrude into the battery box or compartment.

WARNING: When installing or removing a battery, wear safety glasses and take special care to ensure that no sparks are created by tools, or loose jewelry that provide a short to ground. Always remove the ground cable first and install it last. If possible, attach the ground cable to the frame of the battery compartment. Do not lift the battery by their vent tubes, receptacles or terminals.

1016. SUMP JAR. Lead acid batteries are often installed with a sump jar in the exhaust vent that neutralizes vented acid fumes to protect the airframe from corrosive battery acid. If installed, the sump jar should have a capacity of approximately one pint. The jar should contain a 1/2" think pad saturated with a 5% solution of sodium bicarbonate (Baking Soda) in water or about 3/8" of dry sodium bicarbonate. The inlet tube carrying fumes to the sump should extend into the jar about 1" from the lid.

FIGURE 10-2. TYPICAL BATTERY INSTALLATION WITH SUMP JAR



1017. VENTING. Provide suitable venting to the battery compartment to prevent the accumulation of the hydrogen gas expelled during operation. For most aircraft batteries, airflow of 5 cubic feet per minute is sufficient to purge the battery compartment of explosive concentrations of hydrogen:

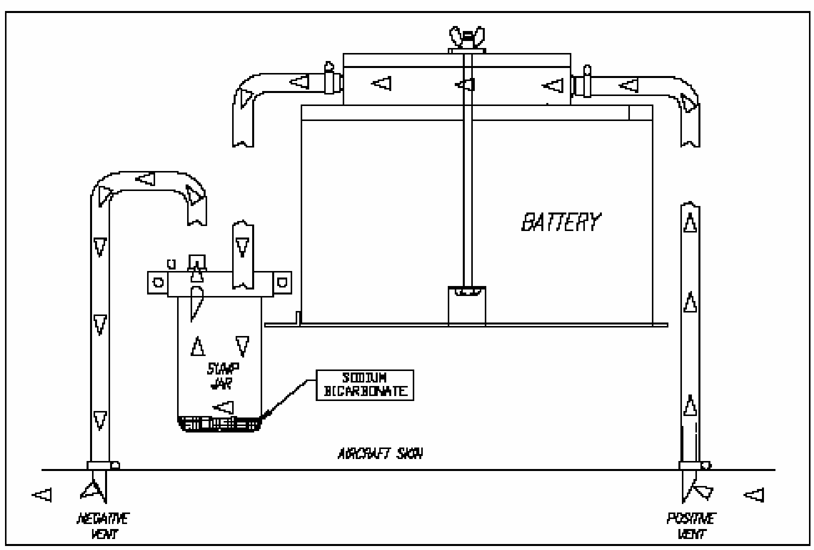
NOTE: Valve regulated lead acid batteries can use the same ventilation system as the original nickel-cadmium battery.

a. Manifold Type. In this type of venting, one or more batteries are connected, via the battery manifold(s) or battery box vents, to a hose or tube manifold system as shown in Figure 10-1. Fasten such hoses securely to prevent shifting and maintain adequate bend radii to prevent kinking.

(1) The upstream side of the system is connected to a positive pressure point on the aircraft, and the downstream side is usually discharged overboard to a negative pressure area. It is advisable to install a battery sump jar in the downstream side to neutralize any corrosive vapors that may be discharged.

(2) When selecting these pressure points, select points that will always provide the proper direction of airflow through the manifold system during all normal operating attitudes. Reversals of flow within the vent system should not be permitted when a battery sump jar is installed, as the neutralizing agent in the jar may contaminate the electrolyte within the battery.

FIGURE 10-3. BATTERY VENTILATION SYSTEM



b. Free Airflow Type. Battery cases or boxes that contain louvers may be installed without an individual vent system, provided that:

(1) The compartment in which the battery is installed has sufficient airflow to prevent the accumulation of explosive mixtures of hydrogen;

(2) Noxious fumes are directed away from occupants; and

(3) Suitable precautions are taken to prevent corrosive battery fluids or vapors from damaging surrounding structure, covering, equipment, control cables, wiring, etc.

1018. DRAINS. Position battery compartment drains so that they do not allow spillage to come in contact with the aircraft during either ground or flight attitudes. Route the drains so they have a positive slope without traps. Drains should be at least 1/2” in diameter to prevent clogging.

1019. ELECTRICAL INSTALLATION.

a. Electrical equipment, controls, and wiring must be installed so that operations of any one unit or system of units will not adversely affect the simultaneous operation of any other electrical unit or system essential to safe operation. Any electrical interference likely to be present in the airplane must not result in hazardous effects upon the airplane or its systems.

b. Cables/Connectors. Use cables and/or connectors that are adequately rated for the current demand and are properly installed. (See AC 43.13-1, Acceptable Methods, Techniques, and Practices-Aircraft Inspection and Repair, chapter 11, as amended.) It may be necessary to contact the battery manufacturer to determine the current value of the battery at the 5-minute discharge rate. Cable size can also be selected by using the same gage as used on a previously approved production aircraft with the same battery.

(1) The cables should be of sufficient length to prevent undue strain on the battery connector or terminals.

(2) Clamp and protect cables, including the bus, in a very secure manner. Since the batteries are not generally fused, any fault in the battery feeder cable could cause loss of the battery electrical system in addition to a possible fire hazard.

(3) Route cables so that cable or terminals cannot short to the battery case or to the hold-down frame.

(4) Route cables outside the battery box whenever practicable to prevent corrosion by acid fumes. When internal routing is unavoidable, protect the cable inside the box with acid-proof tubing.

(5) Assure that cables will not be inadvertently reversed on the battery terminals either by proper cable lengths and clamps or, if this is not practical, use conspicuous color coding.

c. Cable installation must be designed and installed as follows:

(1) Means of permanent identification must be provided for electrical cables, connectors, and terminals.

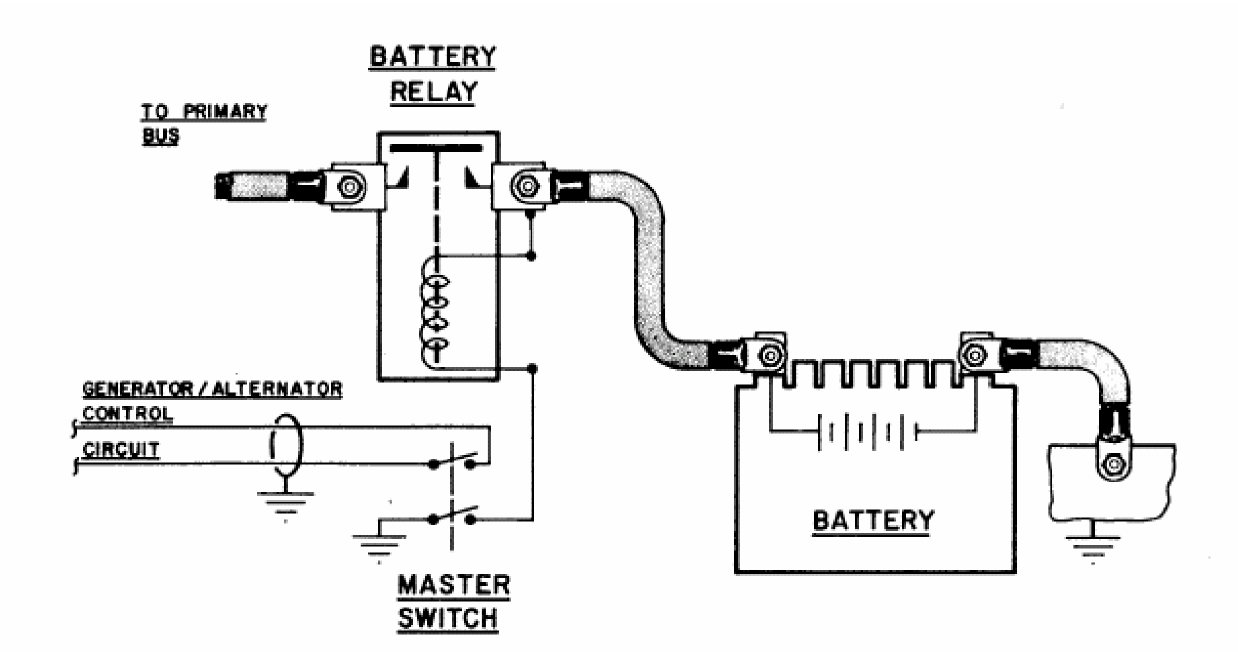
(2) Electrical cables must be installed and secured so that the risk of mechanical damage and/or damage caused by fluids, vapors, or sources of heat, is minimized.

d. When installing lead acid batteries in place of nickel-cadmium batteries some airframes require deactivation or removal of the temperature monitoring systems. Generally this alteration requires a flight manual supplement to override the emergency procedures regarding battery overheating.

NOTE: FAA field approval or STC is required for this kind of alteration.

e. Battery Cut Off. Install a battery cut off relay to provide a means of isolating the battery from the aircraft's electrical system. An acceptable battery cut off circuit is shown in Figure 10-2. Mount the relay so that the cable connecting the relay to the battery is as short as feasible to reduce the possibility of a fire occurring because of a short within this section of cable.

FIGURE 10-4. TYPICAL BATTERY CUTOFF AND GENERATOR ALTERNATOR CONTROL CIRCUIT



1020. QUICK DISCONNECT. Look for and replace the terminal pins when:

a. Excessive pitting or corrosion cannot be removed.

b. Signs of burning or arcing.

c. Cracked part or housing.

d. Excessive wear on contact pins, socket lock pins, or worm screw.

e. Large deposits form on contacts or plastic is discolored.

f. Excessively loose handle and locking assembly.

NOTE: Most manufacturers sell or provide dimensions for tools to check the fit of the quick disconnect sockets.

1021. WEIGHT AND BALANCE. After installation or alteration with the replacement battery the weight and balance of the aircraft should be recomputed if:

a. The weight of the replacement battery is different from that of the original battery.

b. The location of the battery is different from that of the original battery.

NOTE: Weight and balance procedures for aircraft are contained in AC 43.13-1, chapter 10 (as amended).

1022. 1024. RESERVED.

SECTION 3. NICKEL-CADMIUM BATTERY INSTALLATIONS

1025. GENERAL. Nickel-cadmium batteries fulfill a need for a power source that will provide large amounts of current, fast recharge capability, and a high degree of reliability. Nickel-cadmium batteries produce a constant voltage and can operate at lower temperatures. They are generally more expensive to purchase.

1026. ELECTRICAL ANALYSIS. The ampere-hour capacity of a nickel-cadmium battery is selected to accommodate the aircraft load requirements. When making this selection, the following items should be considered:

a. The low internal resistance permits it to recharge very quickly. This high recharge rate can exceed the generator rated capacity and deprive essential circuits of necessary operating current. Total system load (battery recharging plus system loads) must not exceed the pre-established electrical capacity of the generator system.

b. Compare the discharge characteristics curves of the batteries to make sure a reduced capacity nickel-cadmium battery is adequate regarding the following:

(1) Ability to provide engine starting or cranking requirements. Some turbine engines require an initial surge of approximately 1200 amperes, which tapers off within 10 seconds to approximately 800 amperes cranking current. Reciprocating engines require approximately 100 to 200 amperes cranking current.

(2) Ability to provide sufficient capacity for low temperature starting. Nickel-cadmium batteries deliver their rated capacity when the ambient temperature range is 70°F to 90°F. For best engine cranking, replacement batteries with increase capacity will offset the lower power or reduced capacity available when the batteries are cold soaked.

(3) Some nickel-cadmium batteries deliver greater power at temperature extremes than comparable rated lead acid batteries; care must be taken not to overload the cables and connectors.

1027. MAINTENANCE CONSIDERATIONS. To provide for ease of inspection and because nickel-cadmium batteries are generally not serviced in the aircraft, it is important that the battery be located where it can easily be inspected, removed, and installed. Some battery cases are designed with view ports on each side of the case for visual monitoring of the cell electrolyte level. If this type of case is to be utilized, consideration should be given to the location of the battery compartment to accommodate this feature.

1028. STRUCTURAL REQUIREMENTS. Most lead-acid battery compartments provide adequate structure attachment for the installation of nickel-cadmium batteries. However, cantilever supported battery boxes/compartments may not be suitable for nickel-cadmium battery installations unless modified to compensate for an increased over-hang moment. This may be caused by a change in battery shape and center of gravity (c.g.) location even though the replacement battery may weigh less than the original lead-acid battery. Whenever the total installation weight and/or the overhang moment exceed those of the original installation, perform a static test as outlined in chapter 1. If the battery compartment is to be relocated, follow the location requirement procedures outlined in section 1.

1029. ISOLATION OF BATTERY CASE. Because of the material from which nickel-cadmium battery cases are generally made (stainless or epoxy coated steel), it is desirable to electrically isolate the case from the aircraft structure. This will eliminate the potential discharge current produced when spillage or seepage of the electrolyte provides a current path between the cell terminal or connector and the exposed metal of the battery case.

NOTE: Epoxy coated nickel cadmium battery cases serve to

isolate the battery from the airframe thus eliminating electrical leakage to ground. Some batteries use a series of liners that are inside the battery. This isolates the battery from the airframe and helps eliminate electrical leakage to ground.

1030. VENTILATION. During the charging process, nickel-cadmium batteries produce hydrogen and oxygen gases. This occurs near the end of the charging cycle when the battery reaches what is called the gassing potential. To avoid a build up of these gases, and possible accidental ignition, ventilation should be provided to evacuate this gas from the aircraft. There are two types of nickel-cadmium battery cases, one with vent nozzles and one with viewports.

a. The vent nozzle type utilizes vent hoses to evacuate the gas overboard by use of forced air or by venturi effect.

b. Battery cases with viewports or louvers must have airflow sufficient to keep the mixture of air and hydrogen below 4%. The gases from this type of case are evacuated into the battery compartment. Regardless of the ventilation system used, the airflow should be provided at a minimum rate of .040 cubic feet per minute (CFM), this equates to 1.13 liters per minute (lpm).

1031. PRE-INSTALLATION REQUIRE-MENTS. Inspect the replacement battery for possible damage incurred during shipment or storage. Give particular attention to signs of spilled liquid within the shipping container, as it may indicate a damaged cell. Follow procedures outlined in section 2 for battery venting and electrical connections.

a. Pre-installation Battery Servicing. Check the following in accordance with the battery manufacturer's instructions:

(1) Remove the shipping plugs (if used) and clean and install the filler cap vent plugs.

(2) Check the tightness of terminal hardware including each cell connector strap to the proper torque values.

(3) Check the polarity of each cell to be sure they are connected in the proper series or sequence.

(4) Prepare the battery for installation in accordance with the manufacturer's requirements.

b. Compartments or battery boxes which have previously housed lead-acid batteries must be washed out, neutralized with ammonia or a baking soda solution, allowed to dry thoroughly, and painted with alkaline-resistant paint. Remove all traces of sulfuric acid and its corrosive products from the battery vent system to prevent contamination of the potassium hydroxide electrolyte and/or possible damage to the battery case material. Replace those parts of the vent system, which cannot be thoroughly cleansed (hoses, etc.).

1032. NICKEL CADMIUM BATTERY INSTALLATION. Each installation must have provisions to prevent any hazardous effects on structure or essential systems that maybe caused by the maximum amount of heat the battery can generate during a short circuit of the battery or individual cells.

a. Nickel cadmium battery installations must have:

(1) A system to control the charging rate of the battery automatically to prevent battery overheating;

(2) A battery temperature sensing and over-temperature warning system with a means for disconnecting the battery from its charging source in the event of an over-temperature condition; or

(3) A battery failure sensing and warning system with a means for disconnecting the battery from its charging source in the event of battery failure.

b. In the event of a complete loss of the primary electrical power generating system, the battery must be capable of providing at least 30 minutes of electrical power to those loads that are essential to continue safe flights and landing. The 30-minute time period includes the time needed for the pilots to recognize the loss of generated power and take appropriate load shedding action.

1033. SECURING THE BATTERY. Follow the procedures outlined in section 1. Make certain that the hold down bolts are not drawn up too tightly. Ensure that the case to cover seal is installed so that the battery case/cover does not become distorted.

CAUTION: In installations where care has been taken to isolate the battery cases, inadvertent grounding may occur through improper or careless use of safety wire.

1034. QUICK DISCONNECT. Look for and replace the terminal pins when:

a. Excessive pitting or corrosion that cannot be removed.

b. Signs of burning or arcing.

c. Cracked part or housing.

d. Excessive wear on contact pins, socket lock pins, or worm screw.

e. Large deposits form on contacts or plastic is discolored.

f. Excessively loose handle and locking assembly.

NOTE: Most manufacturers sell or provide dimensions for tools to check the fit of the quick disconnect sockets.

1035. VOLTAGE AND CURRENT REGULATION. It is essential that the charging voltage and current be checked and, if necessary, the voltage regulator reset to meet the requirements of the nickel-cadmium battery being installed. IMPORTANT - improper charging current or voltage can destroy a battery in a short period of time.

1036. WEIGHT AND BALANCE. After installation of the nickel-cadmium battery the weight and balance of the aircraft should be recomputed if:

a. The weight of the nickel-cadmium battery is different from that of the original battery.

b. The location of the nickel-cadmium battery is different from that of the original battery.

NOTE: Weight and balance procedures for aircraft are contained in AC 43.13-1, chapter 10 (as amend).

1037. THRU 1040. RESERVED.

SECTION 4. BATTERY INSTALLATION CHECKLIST

1041. STRUCTURAL REQUIREMENTS.

a. Is the battery installed in such a manner that it can withstand the required loads? The effect on other structure (primary or secondary) should be considered.

b. Are suitable materials used in the construction, including standard fasteners, and will the method of fabrication result in a consistently sound structure?

c. If a mounting bracket is used, will the method used in its fabrication produce a consistently sound structure?

d. If the equipment is mounted either on the existing structure or on a bracket attached to the existing structure, is all of the structure (including the bracket, if used) adequate to support the required loads? This answer can be determined by either of two methods:

(1) By direct comparison with an existing approved installation having the same or similar (approximately the same weight and size) equipment installed.

(2) By structural analysis or static test. Such installations do not lend themselves readily to analysis, but are normally adaptable to a static test. In conducting a static test, the following procedure may be used:

(a) Determine the weight and c.g. position of the equipment item.

(b) Mount the equipment in its position in the airplane or simulate the equipment with a dummy so that the required loads can be applied at the c.g. position of the actual equipment.

(c) The required loads should then be applied by any suitable means. If the equipment is light in weight, the inspector could use his own strength and/or weight to determine if the installation will withstand the required loads.

NOTE: All items of mass which would be apt to injure the passengers or crew in the event of a minor crash landing should have their supporting structure designed to the crash load requirements of Title 14 of the Code of Federal Regulations (14 CFR) part 23, § 23.561 or part 25, § 25.561 insofar as the forward, upward, and sideward directions are concerned. The applicable downward load factor shall be the critical flight or landing load factor specified in §§ 23.341 and 23.473, or in § 25.321 or 25.471, whichever is greater.

(3) In lieu of a calculated determination of the down load factor, the ultimate factors of 6.6 and 9.0 may be used for the normal, utility, and acrobatic categories, respectively. For equipment location not covered by § 23.561, the required loads (ref. § 23.301) are the flight and landing load factors of §§ 23.337, 23.341, and 23.473. In lieu of a calculated determination of these loads, the down load factors referenced above may be used.

(4) Supporting structure of other mass items should be designed to the critical flight or landing load factors of § 23.321, 23.471, 25.321, 25.471, 27.321, 27.471, 29.321, or 29.471. The values shown in § 23.561, 25.561, 27.561, or 29.571 may be used in lieu of determination of these values.

e. Is the equipment installed so that it does not adversely affect other structure (either primary or secondary)?

f. Are means provided to permit proper inspections of the installation and related adjacent parts as components?

1042. HAZARDS TO THE AIRCRAFT AND ITS OCCUPANTS.

a. Are the parts of the airplane adjacent to the battery protected against corrosion from any products likely to be emitted by the battery during servicing or flight?

NOTE: Methods which may be used to obtain protection include: acid-proof paint which will resist corrosive action be emitted electrolyte, drain to discharge corrosive liquids clear of the aircraft, positive pressure vents to carry corrosive fumes and flammable gases outside the aircraft, enclosed battery cases which would contain any amount of electrolyte that might be spilled, or combination of these methods.

b. Is the battery container or compartment vented in such a manner that any explosive gases released by the battery during charging or discharging are carried outside the airplane?

c. Is the battery container or compartment vented in such a manner that any noxious gases emitted by the battery are directed away from the crew and passengers?

d. Are the battery connector terminals or other exposed parts protected against electrical contact with the battery container or compartment?

e. Is adequate provision made for the drainage of spilled or excess battery fluid?

1043. OPERATING ASPECTS. If a battery is the only source of electrical power, does the battery have sufficient capacity to supply the electrical power necessary for dependable operation of all electrical equipment essential to the safe operation of the airplane?

1044. DETAIL DESIGN STANDARDS. Is the battery accessible for inspection or servicing on the ground?

1045. RECORDKEEPING.

a. Has a maintenance record entry been made?

b. Has the equipment list and weight and balance been revised?

1046. THRU 1048. RESERVED.

SECTION 5. INSTRUCTIONS FOR CONTINUED AIRWORTHINESS

1049. LEAD-ACID BATTERIES.

a. Airworthiness Limitations.

(1) To ensure continued airworthiness, the battery should be removed and capacity tested. Follow the battery manufacturer's recommended instructions for continued airworthiness (ICA) to determine service periods.

(2) Connector/Wiring Inspection. Check for mechanical integrity, resiliency, pitted, or corroded mating surfaces, burn marks, condition, and type of wiring.

(3) Electrolyte Levels. Electrolyte levels must be maintained just over the plates at all times. Replenish consumed water with distilled or demineralized water.

(4) Sump Jar Maintenance. Inspect the electrolyte levels and the sump jar every 100-flight hours.

b. Capacity Test. The capacity test should be performed as follows:

(1) Check for proper battery installation per STC or manufacturer's ICA when performing annual and 100-hour inspections and when replacing the battery after a capacity test.

(2) Stabilize the battery at 15°C (59°F) or higher. The battery should be at temperature for at least 24 hours.

(3) Remove the battery from the aircraft and charge it according to the recommended charging instructions. Allow the battery to stand on open circuit for one hour.

(4) Connect the fully charged battery to a capacity tester that incorporates a load resistance, amp meter, voltmeter, and a timer.

(5) Discharge the battery at the C1 rate to 1.75 volts per cell (10 volts for a 12 volt battery and 20 volts for a 24 volt battery). Note the discharge time.

(6) The battery is considered airworthy if it meets 80% of its C1 (one hour) capacity rating. However it is recommended to return batteries to service when their capacity is above the minimum; i.e., 85% minimum or 51 minutes to end point voltage.

(7) If the battery fails to meet the minimum run-time, continue by using the constant current (C1) method in the manufacturer's ICA. Allow the battery to stand on open circuit for one hour.

(8) Repeat the discharge test as indicated. If the failure persists, replace the battery.

(9) If the battery is found to be airworthy, it must be recharged with constant potential (CP) prior to re-installing it in the aircraft.

1050. NICKEL-CADMIUM BATTERIES.

a. Airworthiness Limitations. To ensure continued airworthiness, the battery should be removed and inspected per the manufacturer's recommendations.

(1) Connector/Wiring Inspection. Check for mechanical integrity, resiliency, pitted or corroded mating surfaces, burn marks, condition, and type of wiring.

(2) Voltage Regulator. Periodic checks to correct out-of-tolerance regulators and replacement of defective units will reduce the possibility of inadvertent increases in charging voltage with the resultant rise in charge current and battery temperature and water consumption.

(3) Battery. Inspect can and cover for dents, damage, epoxy coating separation, vent tube obstruction, latch function, and cover seal condition.

Remove the battery cover and clean top of cells and connectors with a nylon brush. Verify torque on every intercell connection. If disassembly is required, discharge the battery first.

(4) Electrolyte Levels. Electrolyte levels should be adjusted during the last 15 minutes of the topping charge and while the current is still flowing, because the cells are at their most uniform electrolyte level at this time. Replenish electrolyte levels with distilled, deionized, or demineralized water only. Insure that the proper nozzle and syringe assembly are used to level the cells by referring to the component maintenance manual for the syringe/nozzle specifications. Using the incorrect nozzle may impact battery serviceability and longevity.

(5) Electrical Leakage. Determine if external leakage is of such a magnitude as to require a complete battery cleaning. Follow the manufacturer's recommended procedures.

(6) Sensor Assembly. Inspect the battery for proper placement of thermostats, heaters, thermistors or other sensor elements. Inspect wiring and receptacle for insulation damage, corrosion, and crimping or other defects. If the sensor/harness assembly fails testing or is damaged, it must be replaced. Perform a functional test on the temperature sensor assembly at least once each calendar year.

b. Capacity Test. The capacity test must be performed in accordance with the manufacturer's recommendations.

1051. THRU 1099. RESERVED.

CHAPTER 11. ADDING OR RELOCATING INSTRUMENTS

1100. PURPOSE. This chapter contains structural and design guidance to be considered when aircraft alterations are to be accomplished involving the addition or relocation of instruments.

1101. HAZARDS AND WARNINGS. The rapid advance of technical progress in the aviation industry has resulted in a virtual explosion of high-tech aftermarket instrumentation. In many instances, these innovations are marketed without the support of a proper certification process. As a result, much of the equipment purported to be the most advanced may not be certificated or even certificable for installation on aircraft for which uninformed purchasers have chosen for installation.

WARNING: The burden of provision of and compliance with approved data falls entirely on the technician approving the aircraft for return to service.

1102. ADDITIONAL REFERENCES. Before initiating an alteration involving the addition or relocation of instruments the regulatory basis of the aircraft must be considered. The introduction page of this document provides guidance to be considered on the subject of certification basis. For the purposes of this chapter the following additional references are provided:

a. Department of Commerce, Aeronautics Branch, Aeronautics Bulletin 7-A, section 73(A) through (H) provides regulatory requirements for electrical equipment applicable to the 24 makes and 115 models of aircraft certificated under the provisions of that document. Section 75(B) of the same document provides additional regulatory requirements for instrumentation to be installed on the aircraft listed therein.

NOTE: Care should be exercised in the use of Bulletin 7(A) as certification basis proof data. Subsequent changes in ownership and production of some makes and models may have also resulted in changes of certification basis.

b. Department of Commerce, Civil Aeronautics Administration Bureau of Regulation Inspection Handbook, as revised to January 29, 1947, chapter XVII provides a listing of Approved Type Certificated (ATC) and 609 Group 2 aircraft under those categories. The Group 2 memoranda are identified and briefly described but complete text documentation is not provided. As of this publication no known repository of complete Group 2 memoranda has been identified.

c. Civil Aeronautics Regulation (CAR) 18 and the related Civil Aeronautics Manual (CAM) 18 established and maintained the standards and practices for all maintenance, repair, and alteration of aircraft applicable under the provisions of the CAR. That document remained in effect until the advent of the systemic changes dictated by the Federal Aviation Act of 1958 and the resultant documentary changes.

1103. PREPARATION. Before initiating any alteration involving the addition or relocation of instruments the regulatory basis of the aircraft must be considered. Chapter (*Un-numbered)* provides guidance to be considered on the subject of Certification Basis. For the purposes of this chapter the following references are provided:

- Civil Air Regulation (CAR) 3.661 thru 3.676 provide the regulator basis for installation of instruments CAR 3 airplanes
- CAR 4a532 thru 4a537 provide the regulatory basis for supplemental variations for installation of instruments

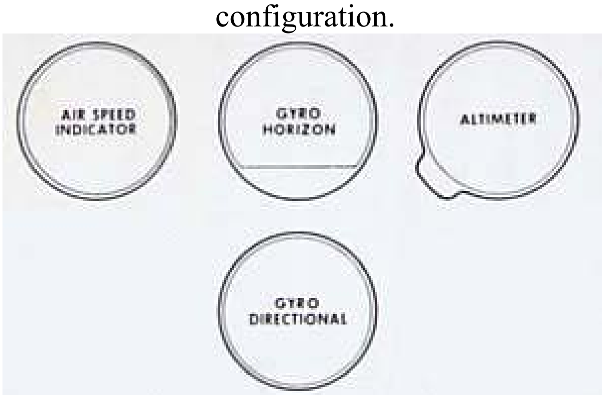
in CAR 4 non-air carrier (NAC) airplanes

- Title 14 of the Code of Federal Regulations (14 CFR) part 23, § 23.1321 provides the regulatory basis for the arrangement and visibility of instruments in part 23 airplanes

a. Structure. Chapter 1, Structural Data, provides guidance by which structural integrity may be determined. Chapter 2, paragraphs 202 thru 203 provide information pertinent to instrument panel installations may be determined.

b. Location. In the absence of specific regulatory requirements, installation of instruments required for operation under Instrument Meteorological Conditions (IMC) the recommended configuration is in the form of the basic "T" (Plate 11.1).

Plate 11.1: The basic "T" VMC instrument configuration.



(1) The aircraft Attitude Indicator is located at the top center of the installation.

(2) The Air Speed Indicator is located adjacent to and left of the Attitude Indicator.

(3) The Altimeter is located adjacent to and to the right of the Attitude Indicator.

(4) The Directional Indicator is located adjacent to and immediately below the Attitude Indicator.

NOTE: As instrument panel installations have become more complex the basic "T" has become more difficult to accomplish and, with the relocation of the Directional Indicator, the relationship of the ASI-ATT-AI combination is assumed.

1104. INSTALLATION. Throughout the development of the complexity of aircraft the requirements for the mounting of all instruments has remained the same. All instruments shall be mounted so they are visible to the crewmember primarily responsible for their use.

a. Structure. Prior to initiating changes to instrument panel installations it is necessary to determine the load bearing requirements of the component. In some aircraft the instrument panel is stressed as the primary structure while the majority is treated as the secondary structure with load bearing responsibilities limited to instruments and equipment installed. Regardless of their structural nature it is the responsibility of the altering technician to return the installation to its original or properly altered state.

(1) In all cases, where it is available, it is necessary to refer to the manufacturer's instructions for continued airworthiness when considering alterations to any structure. In the case of aircraft manufactured before the regulatory requirement for the manufacture to provide such information reference should be made to AC 43.13-1, Acceptable Methods, Techniques, and Practices, chapter 4, section 4 (as amended) and/or AC 43.13-2B, chapter 2 for methods and techniques of retaining structural integrity.

(2) Failing availability or applicability of those sources it is recommended for the altering individual to avail her/him selves of the services of a designated engineering representative (DER) to obtain approval of the data prior to accomplishment of the task.

b. Instrument Plumbing. The majority of CAR 3 and CAR 4 general aviation aircraft utilize instrumentation that are identified as "wet" gauges. This term has nothing to do with what is being measured by the gauge but rather the material to be measured is delivered directly to the gauge. Typical "Wet" systems are fuel pressure, engine oil pressure and, when so equipped, hydraulic pressure. While not usually considered "wet" systems the airspeed and vacuum systems use this same system. The tubing used is generally thin walled seamless aluminum or stainless steel. However, flexible rubber and/or neoprene lines are prevalent in airspeed and vacuum sensing lines.

(1) The most reliable source of information on the material used in a particular aircraft is the manufacturer's ICA. If that source is not available, AC 43.13-1, chapter 8 (as amended) provides inspection and maintenance guidance on fuel lines while chapter 9 provides the same level of information on hydraulic systems.

(2) With the advent of part 23, the installation of remote sensing systems have served to replace the "wet" systems with system pressures being remotely measured by transducers and electrically transmitted to the cockpit indicators.

(3) Aircraft with long histories of alteration may contain a mixture of mechanical and/or electrical analogue instruments and digital presentations.

c. Venturi Vacuum Source. A venturi is a relatively low cost means of producing the vacuum to operate gyroscopic instruments such as the turn and bank, directional gyro and artificial horizon. It is mounted on the exterior of the fuselage, parallel with the longitudinal axis of the aircraft. As the aircraft moves through the air an area of reduced pressure, a partial vacuum, is created in the throat of the venturi. The venturi throat is connected by a tube to the gyro instrument case resulting in a reduction of pressure within the case. Ambient air, entering the instrument through the inlet port of the case passes over the gyro rotor causing it to spin rapidly and causes the instrument to activate. Venturis are available in sizes providing 2", 4" and 9" Hg. of vacuum.

(1) The vacuum requirement for the operation of each instrument is generally provided on the data plate attached to its case. If it is not then it is necessary to obtain the data from the manufacturer or another acceptable source such as AN and MS standards.

(2) In the event the vacuum available exceeds the requirements of the installation it will be necessary to install an in-line regulator to adjust the flow within requirements.

d. Pumped Vacuum Source. Current aviation technology has produced durable light weight, mechanically driven, pump and control systems for providing instrument operating vacuum. Selection of the correct pump remains dependent on the vacuum requirements of the system being driven. Control is provided by either the system itself or an in-line regulator. Production and aftermarket standby Vacuum systems are available to provide operating capabilities in the event of failure of the primary mechanically driven system.

NOTE 1: For the purposes of this document this text provides an example problem. Statement of the problem given the data provided herein for the equipment listed, determine the minimum pressure flow and vacuum to provide for operation of the example unit(s) listed.

e. Calculating Vacuum Loads. When a venturi vacuum source is selected it should not be taken for granted that the venturi selected will provide sufficient flow and negative pressure to operate the instrument package selected. Even within the make and model of the venturi selected there may be sufficient tolerance to fail to meet the requirements of the system selected. Therefore, it is essential that the vacuum load requirements be carefully evaluated.

(1) Gyroscopic instruments require optimum value of airflow to produce their rated rotor speed. For instance, a specific bank and pitch indicator required approximately 2.30 cubic feet per minute flow and a resistance, or pressure drop, of 4" Hg. If the altering technician has selected a 2" Hg. Venture, the resultant vacuum would be insufficient to drive the instrument.

(2) For the purposes of this document the following vacuum driven items have been selected as examples for the purposes of calculation:

TABLE 11-1. CFM/VACUUM VALUES

instrument	*flow*	*vacuum*
Bank and Pitch Indicator	2.3 cubic feet per minute	4.0" Hg
Directional Gyro Indicator	1.3 cubic feet per minute	4.0" Hg
Turn and Bank Indicator	.5 cubic feet per minute	2.0" Hg
Total flow required	4.10 cubic feet per minute	

The above listed instruments are listed in Tables 11-2 and 11-4. Optimum values are shown in Table 11-4. It should be noted that the negative pressure air source must deliver not only the optimum value of vacuum for the instruments, but must also have sufficient volume capacity to accommodate the total flow requirements of the various instruments it serves.

NOTE 2: For the purposes of this text the example components listed above and in Tables 11-2 and 11-4 are accepted as viable and the operating parameters related to them as accurate. The nomenclature and the properties related have no effect on the resulting computations.

NOTE 3: For the purposes of this text three example tables are provided.

TABLE 11-2. PROVIDES INSTRUMENT AIR CONSUMPTION FOR AN EXAMPLE VACUUM DRIVEN AUTOPILOT AND INSTRUMENT INSTALLATION

Instrument	Air consumption at sea level	
	Differential drop in. Hg suction (Optimum)	Cubic feet/per minute
Automatic Pilot System (Type A-2, A-3, & A-3A)		
Directional gyro control unit across mount assembly	5.00	2.15*
Bank & climb gyro control unit across mount assembly	5.00	3.85*
Total	—	6.00*
Automatic Pilot System (Type A-4)		
Directional gyro control unit	5.00	3.50*
Bank & climb gyro control unit	5.00	5.00*
Total	—	8.50*
Bank & pitch indicator	4.00	2.30
Directional gyro indicator	4.00	1.30
Turn & bank indicator	2.00	.50
(*) NOTE.— Includes air required for operations of pneumatic relays.		

TABLE 11-3. PROVIDES THE EQUIVALENT STRAIGHT TUBE LINE DROPS FOR THE 90 DEGREES ELBOWS INSTALLED IN ANY GIVEN SYSTEM (BY O.D.)

Tubing size			Pressure drop in a 90˚ elbow in terms of length of straight tube equivalent to a 90˚ elbow
O.D inch		Wall thickness inch	Feet
1/4	X	.035	0.28
3/8	X	.035	0.46
1/2	X	.042	0.62
5/8	X	.042	0.81
3/4	X	.049	0.98
1	X	.049	1.35

TABLE 11-4. PROVIDES THE DIFFERENTIAL PRESSURE ACROSS THE INSTRUMENT INLET AND OUTLET OF THE EXAMPLE INSTRUMENT PACKAGE

Instrument	Suction in inches of Mercury		
	Minimum	Optimum	Maximum
Automatic Pilot System (Type A-2, A-3, & A-3A)			
Directional gyro control unit across mount assembly	4.75	5.00	5.25
Bank & climb gyro control unit across mount assembly	4.75	5.00	5.25
Gauge reading (differential gauge in B & C control unit)	3.75	4.00	4.25
Automatic Pilot System (Type A-4)			
Directional gyro control unit	3.75	5.00	5.00
Bank & climb gyro control unit	3.75	5.00	5.00
Bank & pitch indicator	3.75	4.00	5.00
Directional gyro indicator	3.75	4.00	5.00
Turn & bank indicator	1.80	2.00	2.20

(3) Calculation of equivalent straight line pressure drop due to routing. In addition to the effects of the instrumentation itself consideration must be given to the parasitic resistance to flow provided by the routing of the tubing providing the source pressure to the instruments. For the purposes of this problem it is assumed the example system has 4 right angle elbows and 20 feet of 1/2 inch O.D. X .042" tubing.

(4) Solution to the stated problem (step-by-step).

(a) Accept the flow requirements of the example instruments listed in paragraph 213e(2) as 4.10 CFM. The pressure drop for one 90-degree 1/2 inch O.D. elbow is equivalent to 0.62 feet of straight tubing. Therefore, the pressure drop of the four 90-degree elbows is the equivalent to 2.48 feet of tubing.

(b) Determine the pressure drop through 22.48 feet (20 feet plus the above 2.48 feet) of 1/2 inch O.D. X .42 inch tubing at 4.10 CFM flow.

1. The pressure drop for each 10 foot length of that tube is .68" Hg.

2. Divide 22.48 feet of tubing to determine the number of ten foot lengths (i.e., 22.48 divided by 10 equals 2.248).

3. Multiply the number of sections (2.248) by .68" Hg to obtain the pressure drop through the system (.68 X 2.248= 1.53" Hg).

4. The pump must therefore be capable of producing a minimum pressure differential of 5.63" Hg (i.e., 4.10" Hg) for maximum instrument usage + 1.53" Hg (determined) at a flow of 4.10 CFM.

(c) Filters are used to prevent dust, lint, or other foreign matter from entering the instrument and vacuum system. Filters may be located at the instrument intake port or the manifold intake port when instruments are interconnected. It must be determined that the flow capacity of the filter is equal to or greater than the flow capacity of the vacuum system. If it is not, the restriction will create a pressure drop in the system. AC 43.13-1, chapter 12, paragraph 38a through d (as amended) provides additional guidance on the installation and maintenance of venturi and engine driven vacuum and instrument systems.

(d) Electrical supply for instruments. AC 43.13-1, chapter 11 (as amended) provides sixteen sections of text on the subjects of inspection and installation of electrical systems.

(e) Instrument lighting. The regulatory requirements for instrument lighting have changed little as certification specifications have developed.

1. CAR 3.696 and 3.697 provide two paragraphs simply stating (1) instrument light installations should be safe, (2) provide sufficient illumination to make instruments and controls easily readable, and (3) shall be installed in such a manner as not to shine in the pilot's eyes.

2. CAR 4.a.577 adds that instrument lighting shall be equipped with a rheostat control for dimming unless it can be shown a non-dimming light is satisfactory.

3. Section 23.1381 consolidates the language of the earlier regulatory requirements with no appreciable changes.

(f) Magnetic Direction Indicator (magnetic compass). The maximum deviation limitations of magnetic compasses have changed little since 1928.

1. CAR 3.666 specifies the Magnetic Direction Indicator shall be installed that its accuracy shall not be excessively affected by the airplane's vibration or magnetic field. After compensation the installation shall be such the deviation level in flight does not exceed 10 degrees on any heading. CAR 3.758 provides and requires the installation and maintenance of the Compass Deviation Card.

2. CAR 4.a.562 provides language that refers to dampening and compensation of the installation as well as acknowledgement of the effects of electrical disturbances resulting from the increase in proliferation of aircraft electrical systems. As a result, specific aircraft required to meet the provisions of CAR 4a may be required to display two compass correction cards, one for operation with electrical power off, and one for electrical power on operation.

3. Section 23.1327 provides location and deviation considerations also provided for under CAR 3.666 and CAR 4.a.562. This section additionally provides for more than 10 degrees deviation of the non-stabilized magnetic compass due to the operation of adjacent high draw electrical components: if either a magnetic stabilized indicator, not having a deviation of greater than 10 degrees on any heading, or a gyroscopic direction indicator is installed.

(g) Alternative installations to liquid filled Magnetic Direction Indicators. With the development of the vertical reading dry compass it has become practical to replace the time honored liquid filled compass with a viable and approved alternative.

1. Vertical reading dry compasses provide the pilot with a true relationship of the aircraft to azimuths as opposed to the reverse reading requirements of the liquid filled compass.

2. The vertical reading dry compass does not leak and required no periodic refilling or cleaning to make it stable and readable.

3. The vertical reading compass provides no relief from the requirement for magnetic correction compensation. Depending on the manufacturer and individual installation, the unit may even require supplemental compensation to meet the maximum deviation requirements.

4. Vertical reading compasses are available on the aftermarket in both TSO and not-TSO versions. The altering technician must be aware of the limitations and certification responsibilities of the unit selected.

1105. TESTING, MARKING, AND PLACARDING.

a. Testing the Venturi Tube Powered Systems. At normal in-flight cruise speed, or an accurately generated representative airflow, check the venture tube-powered system to assure that the required vacuum is being supplied to the system.

b. Testing the Vacuum Pump Powered System. When the system is powered by either engine or auxiliary driven vacuum pumps, check the system for proper output at their rated RPM. The output should be measured at the point of delivery to the instrument.

c. Testing of Altimeters and Static Systems. Prior to performing an altimeter or static system test, determine that it is free of contaminating materials such as dirt and water. With all instruments disconnected from the system, purge the plumbing with low pressure dry filtered air or nitrogen. Part 43, appendix E provides guidance and practical guidance on altimeter and static systems tests and inspections.

NOTE 4: Altimeter tests in accordance with appendix e must be accomplished by a properly certificated repair agency.

d. Testing Instrument Electrical Supply. Subsequent to major repairs or alterations resulting in addition, replacement or rearrangement of instrument electrical circuitry, it is necessary to verify continuity and current availability prior to, connection of the instruments. With the individual instruments isolated from the power circuits determine if the current available is in accordance with the manufacturer's requirements.

e. Fuel, Oil, and Hydraulic Fluid Supply (Wet Instruments). It is necessary to verify that the fluid transmission lines are free of residual material, dirt, and water prior to their connection to the instruments. Purging should be accomplished by low pressure application of dry filtered air or nitrogen.

(1) On many CAR 3 and CAR 4a aircraft common sized fluid connections provide the opportunity for erroneous connection of incorrect fluid lines to the wrong instruments. Care should be exercised to assure source and delivery of the required material to the correct instrument.

(2) Fluid pressures should be verified at both the source and instrument to verify the absence of restrictions or blockage before connection to the instrument.

f. Fuel, Oil, and Hydraulic Fluid Supply (Dry Instruments). The same care exercised on the verification of security, source, and hygiene of wet instrument systems should be rendered to dry systems.

(1) The primary difference is the delivery of the material to be measured to a transducer or pressure transmitter rather to the back of an instrument. The wrong product delivered to the incorrect instrument can result in its damage beyond repair.

(2) The additional difference between the two instrument systems is the attention required to provide the correct electrical circuitry and current between the transmitter and the instrument. (See subparagraph d.)

g. Instrument Markings and Placards.

(1) When additional instruments are installed they must be appropriately marked. Refer to the applicable CAR and 14 CFR for specific instrument marking and placard requirements.

(2) Particularly in cases where the instrument panel was replaced it is necessary for the altering individual to refer to NOTE 2 of the applicable ATC, TC, Group 2, Bulletin 7A, or TCDS of the aircraft to assure proper installation and placement of the required placards and limitations.

1106. ELECTRONIC DISPLAY INSTRUMENT SYSTEMS OR ELECTRONIC DISPLAY INDICATORS. The installation of these systems, regardless of the certification basis of the airplane into which they are being installed, are defined and specified under the provisions of § 23.1311.

a. Electrostatic Discharges.

(1) With new technologies come the requirements to learn new skills. While a certain amount of care and attention to detail while handling mechanical and electrical instruments has always been required, the hazards have been based on circumstances that could be seen and felt. With the glass cockpit a different kind of hazard has surfaced as a real threat to airworthiness of instrumentation. That threat is called Electrostatic Discharges (ESD).

(2) ESD is not a new phenomena. It has been with the industry a long time. However, with the advent of the electronic display instrument system and its multiple presentations, the problem can now disrupt entire display systems as a result of one event. Any unit containing electronic components such as diodes, transistors, integrated circuits, programmable Read-only memory (ROM), ROM, and memory devices must be protected from ESD. The simple act of improperly carrying an electronic device across a room without adequate protection can render it unairworthy.

b. Electronic Display Instrument Systems Provide no Relief from the "Basic T". Section 23.1311(a)(5) requires systems utilizing electronic displays to have independent backup systems to provide basic VMC reference in the event of loss of electrical power.

(1) Subparagraph 5 requires an independent magnetic indicator and either an independent secondary mechanical altimeter, airspeed indicator and attitude instrument; or

(2) An individual electronic display indicators for altitude, airspeed, and attitude that are independent from the airplanes primary electrical system.

(3) These secondary instruments may be installed in panel positions that are displaced from the primary positions as specified by § 23.1321(d) but must be located where they meet the pilot's visibility requirements of § 23.1321(a).

CAUTION: To prevent damage due to excessive electrostatic discharge, proper gloves, finger cots, or grounding bracelets should be used. Observe the standard procedures for handling equipment containing electrostatic sensitive devices or assemblies in accordance with the recommendations set forth in the maintenance instructions published by the manufacturer.

c. AC 23.1311-1, Installation of Electronic Display in Part 23 Airplanes (as amended), paragraph 3.0, provides valuable interface references relating to the relationships between the CAR and 14 CFR regulatory requirements installation of this technology to older and current technology in airplanes.

1107. ENVIRONMENTAL CONDITIONS.

a. General. The equipment environmental limits established by the manufacturer should be compatible with the operating environment of the airplane. Evaluation of the equipment installation should consider such factors as the maximum operating altitude of the airplane and whether the equipment is located within a temperature and pressure controlled area. Applicable methods for testing the performance characteristics of the equipment for specific environmental conditions are provided in RTCA/DO-160( ). Either test or analysis, or both, ensures the compatibility between the operational environment and the environmental equipment category of the laboratory tests.

b. Temperature. Electronic systems reliability is strongly related to the temperature of the solid-state components in the system. Component temperatures are dependent upon the internal thermal design and external cooling. In evaluating the temperature requirements, consider the additional heat generated by the equipment, especially in an area where air flow is restricted. To determine if adequate cooling is provided, the evaluation must make maximum use of previous data from compatible installations. This will assist in limiting the ground and/or flight tests of those installations that cannot be verified by other means. When the equipment operating environment can not be verified from previous experience or from evaluation of temperature values in that equipment location, a cooling test must be conducted.

c. Attitude Information. Attitude information should continue to be presented for a minimum of 30 minutes after the in-flight loss of cooling for the primary instrument when in the normal operating environment (temperature/altitude). If proper performance of the flight instrument functions is adversely affected due to loss of in-flight cooling, such conditions must be enunciated. Consideration should be given to incorporation of an over-temperature shut-down of the system in the vent of cooling system failure. In the event of inclusion of such systems, AFM documentation should be established requiring subsequent pilot actions. Additionally, applicable placards must be provided for pilot situational awareness of the critical condition. These actions should include procedures to allow possible recovery of a system that has had an over-temperature shutdown condition.

d. Enunciation. Enunciation of in-flight loss of cooling or fan monitors may not be required if shown by a safety analysis or test demonstration that a hazardous or catastrophic condition is not indicated. The safety analysis should consider the reliability of the fans, redundancies of functions, reversionary features (such as the ability to transfer critical functions), the enunciation of over-temperature and its response time, and the availability of other flight instrumentation. In some systems, cooling fans may be installed to improve the operating environment of the components and reduce the possibility of a failure condition or shutdown of the equipment. In the event of those supplementary installations, fan monitoring or additional temperature sensors may not be required. If cooling fans are needed to prevent a hazardous or catastrophic failure condition, installation of fan monitoring or other methods of determining the status of the cooling fan must be provided prior to flight.

1108. THRU 1199. RESERVED

CHAPTER 12. CARGO TIEDOWN DEVICE INSTALLATIONS

1200. PURPOSE. This chapter provides data for making acceptable cargo tiedown device installations in non-pressurized areas of civil aircraft of 12,500 lbs gross weight or less. Engineering assistance may be required for floor/attach fittings load analysis and material burn testing.

1201. HAZARDS/WARNINGS.

a. Structural failure may occur if aircraft floor loading limits are exceeded.

b. Fire hazards may exist if materials do not meet flame resistant specifications.

c. Exceeding manufacture weight and balance limitations create unsafe flight conditions.

1202. INSTALLATION CONSIDERATIONS.

a. Assure that the altered aircraft can be operated within the permissible weight and center of gravity (c.g.) ranges.

b. Determine that there will be free access to all equipment and controls essential to the proper operation of the aircraft, required emergency exits, and emergency equipment.

c. Use only materials that are at least flame resistant for covering of floors and webbing material. Refer to the applicable airworthiness standards for the aircraft involved to determine the required flame-resistant qualities. For aircraft in air taxi or other commercial operations, refer to the applicable operating rule for special requirements regarding fire protection, cargo bins, location of cargo with respect to passengers, cargo compartment, or aisle width.

1203. FABRICATION AND INSTALLATION.

a. Cargo Tiedown Devices.

(1) Cargo tiedown devices may be assembled from webbing, nets, rope, cables, fittings, or other material which conforms to a FAA-PMA, TSO, NAS, AN or MIL-SPEC standards. Use snaps, hooks, clamps, buckles, or other acceptable fasteners rather than relying upon knots for securing cargo. Install tensioning devices or other acceptable means to provide a method of tightening and adjusting the restraint system to fit the cargoes to be carried.

(2) Provide covers or guards where necessary to prevent damage to or jamming of the aircraft's equipment, structure, or control cables.

(3) Straps and nets manufactured with Military Specification webbing and thread must be evaluated to determine that the working load requirements are met. All tiedown assemblies are as strong as the weakest component in the system including the point of attachment.

NOTE: The owner/operator is responsible to ensure that a basis of approval/acceptance is obtained for the tiedown device before use on aircraft. This is accomplished by providing substantiating engineering data of the devices being used, including aircraft floor load limits.

b. Structural Attachment. Commercially available seat tracks, rails, or other types of anchor plates may be used for structural attachment, provided they conform to conforms to a NAS, TSO, AN or MIL-SPEC standard. This type of hardware permits a ready means of mounting a wide variety of quick-disconnect fittings for cargo tiedown. Typical examples of such fitting and their attachment are shown in Figures 12-1 through 12-5.

(1) When installing these fittings, reinforce the existing floorboards and/or other adjacent structure to obtain the necessary load carrying capacity. Seat tracks installed longitudinally across lateral floor beams generally require full-length support for adequate strength and rigidity between beam attach points (see Figure 12-4).

(2) Consider the inherent flexibility of the aircraft structure and install any reinforcement in a manner that will avoid localized stress concentrations in the structural members/areas. Give specific attention to the size, shape, and thickness of the reinforcement, fastener size and pattern, and the effects of any adhesives used.

(3) Fittings used for cargo tiedown attachment need not be substantiated by static tests if it can be shown that the fitting's rated minimum breaking strength would not be exceeded by the applicable static test loads. Existing racks, rails, or other points used for attachment may be verified by static tests, analysis, or a written statement by the aircraft manufacturer attesting to its adequacy to withstand the necessary loads.

FIGURE 12-1. EXTRUDED TRACK, ANCHOR PLATES, AND ASSOCIATED FITTINGS

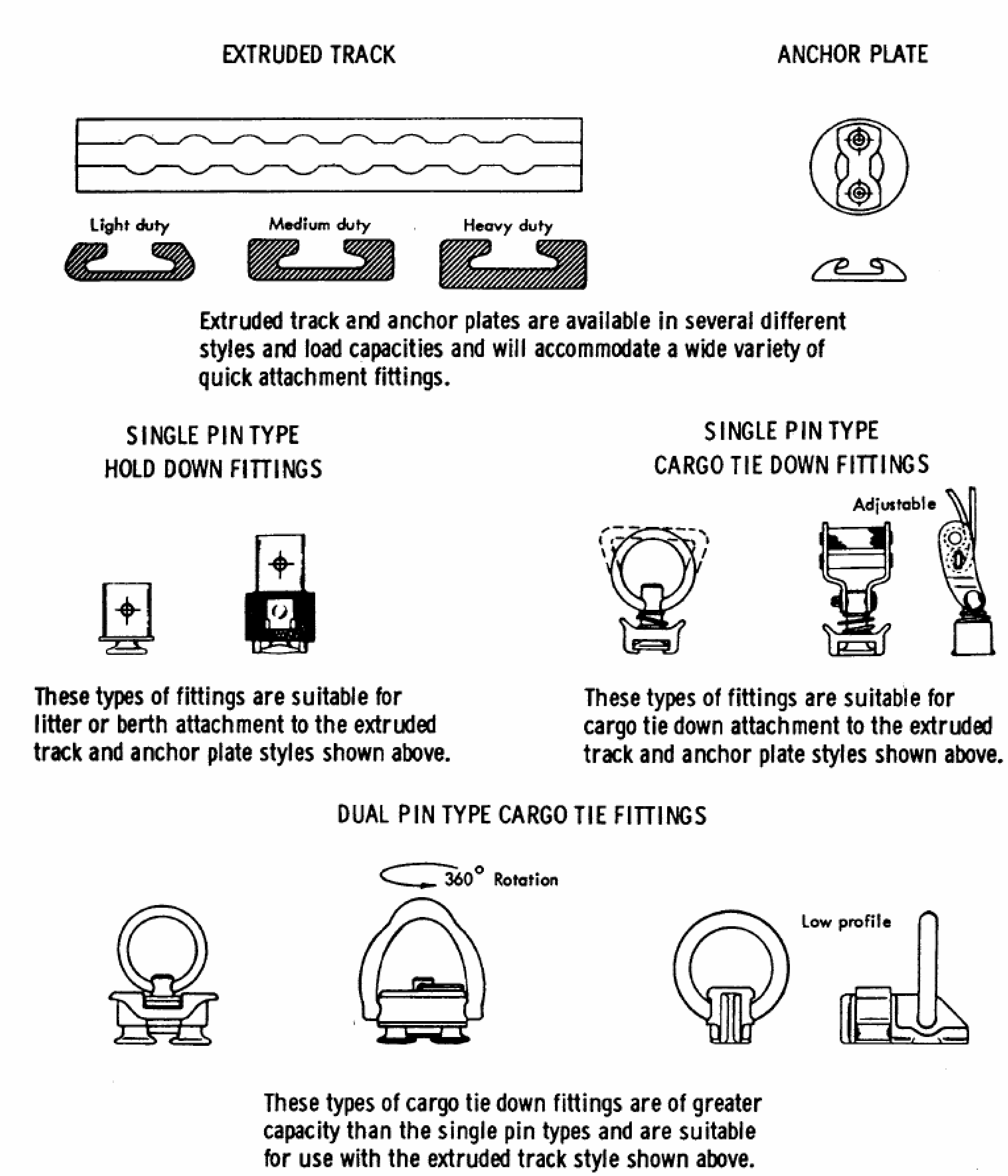


FIGURE 12-2. MISCELLANEOUS LITTER, BERTH, AND CARGO TIEDOWN FITTINGS

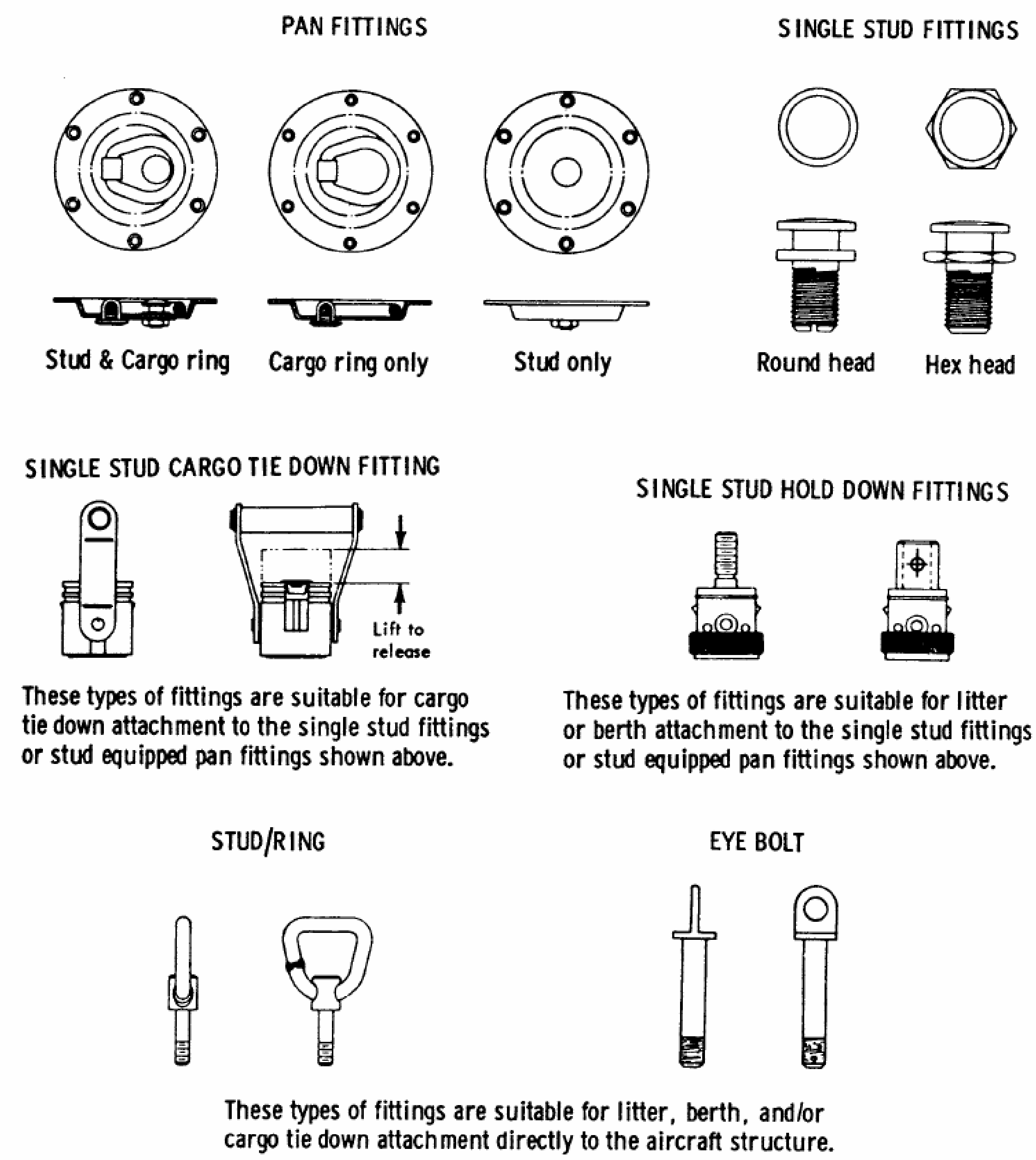
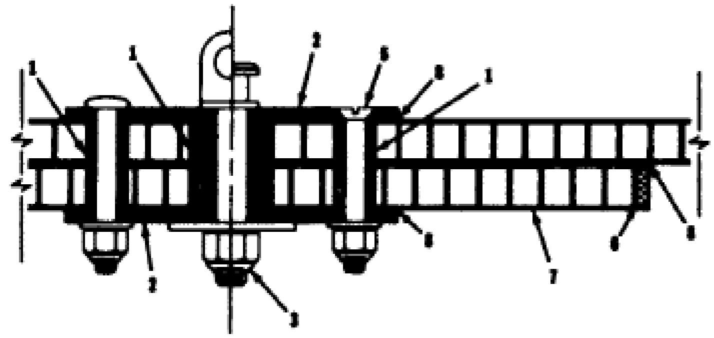
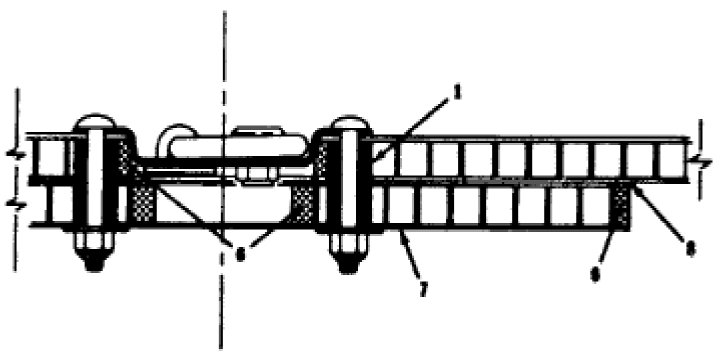


FIGURE 12-3. TYPICAL ATTACHMENT OF FITTINGS TO HONEYCOMB STRUCTURES

A. Attachment method utilizing a honeycomb doubler.

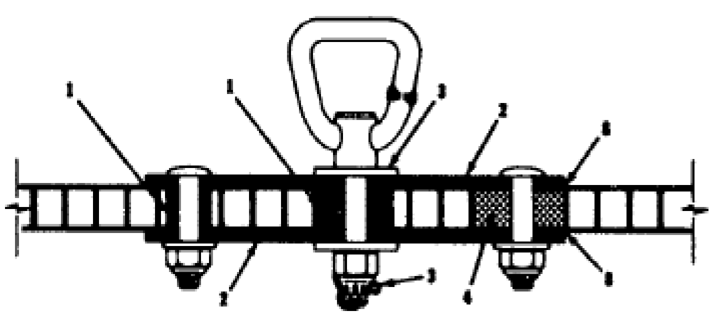


Single Studs or Eye Bolts.



Pan Fittings.

B. Attachment methods utilizing reinforcing plates.



Stud/Rings.

FIGURE 12-3. TYPICAL ATTACHMENT OF FITTINGS TO HONEYCOMB STRUCTURES – CONTINUED

1. Bed all inserts and spacers in a suitable potting compound.
2. Reinforcing plate.
3. Where fitting is subject to rotation, place washers on both sides and use a positive safety means.
4. (Alternate method in lieu of spacers) Undercut honeycomb, inject potting compound, and drill through when set.
5. Countersink if required for clearance or if desired for appearance.
6. Undercut all open edges of honeycomb 1/16" and seal with potting compound.
7. Honeycomb doubler.
8. Use epoxy or other suitable adhesive to attach doubler and reinforcing plates.

FIGURE 12-4. INSTALLATION OF UNDERFLOOR SUPPORT FOR EXTRUDED TRACK

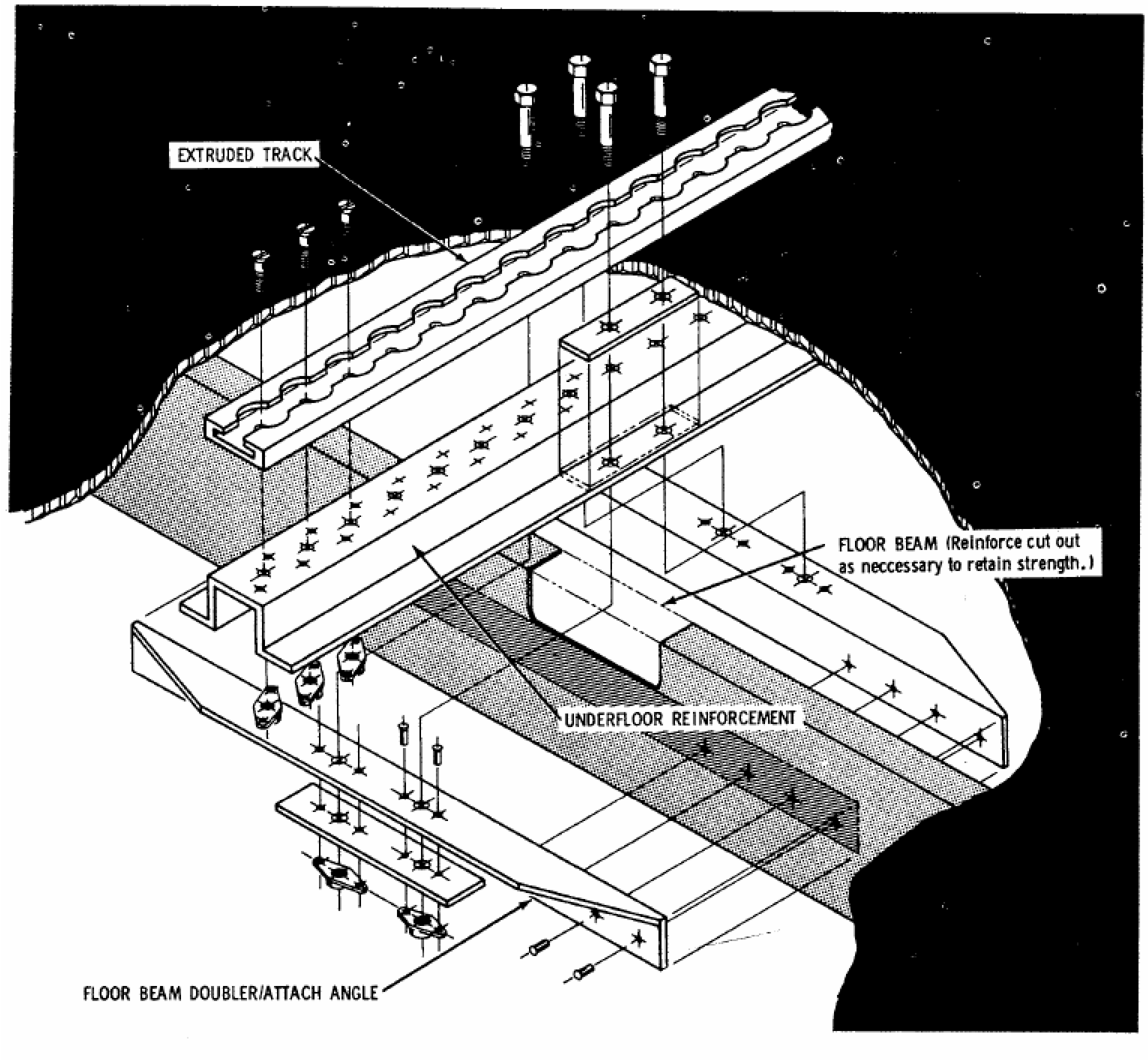
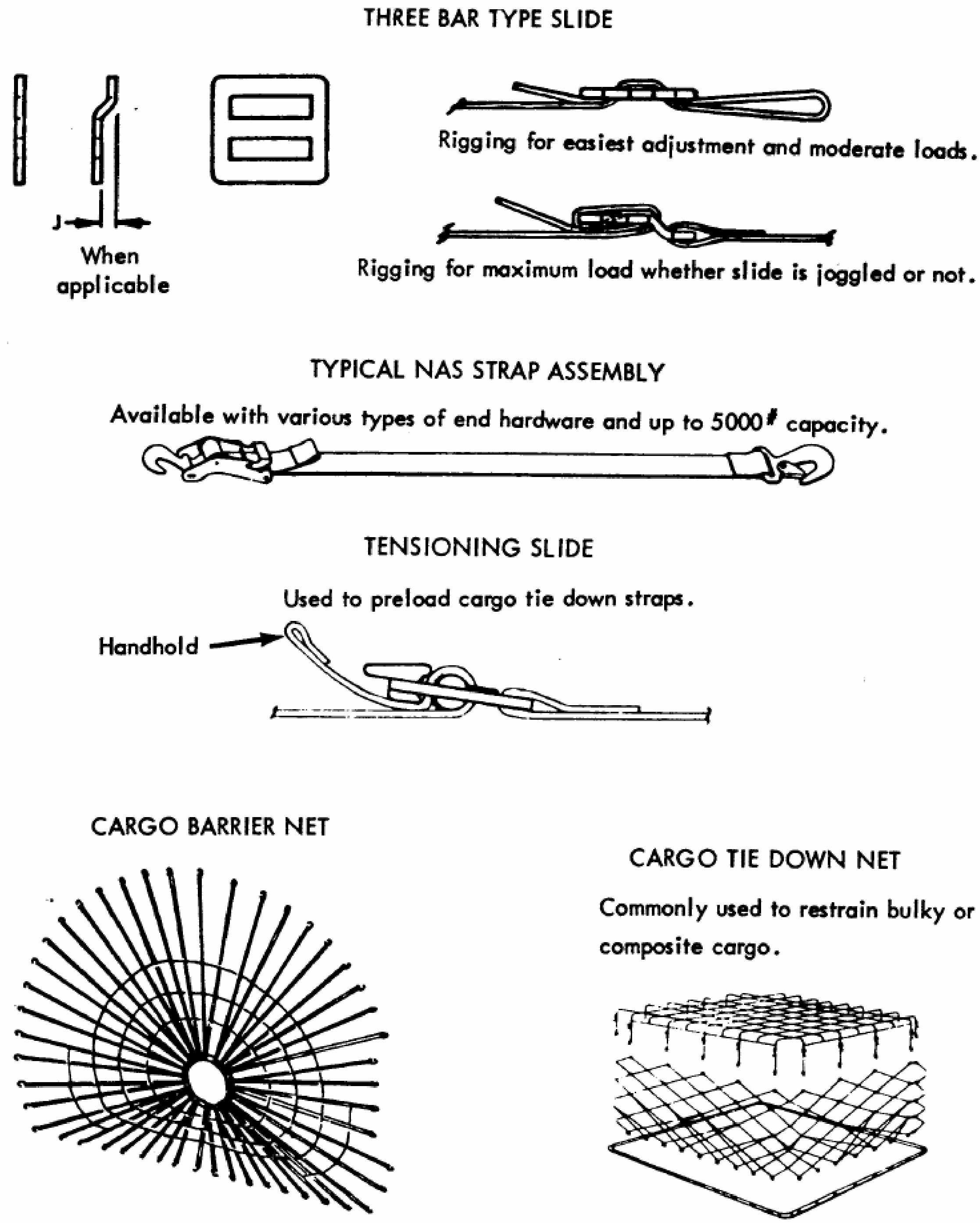


FIGURE 12-5. TYPICAL CARGO TIEDOWN STRAPS AND CARGO NETS



c. Load Factors.

(1) Use the load factor established by the aircraft manufacturer for type certification as the basis for substantiating the cargo tiedown devices and attachments to the aircraft structure. Refer to the applicable operating rules for any additional load factor requirements if the aircraft is to be used for air taxi or other commercial operations.

(2) The critical load factors to which the installation is to be substantiated are generally available from the holder of the aircraft's Type Certificate (TC). When the TC holder is no longer active, such data may be obtained from the controlling FAA regional engineering office.

(3) The maximum loadings are a requirement of the aircraft manufacturer and must be adhered to. Aircraft manufacture floor loading limitations take precedence over the load rating of the net, fitting, and tiedown devices.

d. Static Tests.

(1) It is recommended that static testing be conducted on a duplicate installation in a jig or mockup that simulates the related aircraft structure. Refer to chapter 1, paragraph 3 for static test information.

(2) If the actual installation is used for static testing, inspect both the aircraft and the cargo tiedown device installation thoroughly before releasing to service. Check all members and fittings for cracks, distortion, wrinkles, or elongated holes. Replace all bolts and threaded fittings that are not inspected by magnetic particle or other acceptable N.D.T. inspection process. Inspect riveted joints for tipped rivet heads and other indications of partially sheared rivets or elongated holes.

(3) All cargo tiedown installations must be tested to the critical ultimate load factor. Refer to chapter 1 of this handbook for computation and testing procedures.

(4) When the cargo compartment is separated from the cockpit by a bulkhead that is capable of withstanding the inertia forces of emergency conditions a forward load factor of 4.5 g may be used. All other applications require the use of a 9 g forward load factor.

(5) Each cargo tiedown fitting installation must be static tested under forward, side, and up load conditions. Individual fittings may be tested by applying a single pull of 12.6 g forward load at an angle of 18.5° up and 9.5° to the left or right (as applicable) of the aircraft longitudinal axis. For example, assuming a 5,000-lb static pull (rating of a typical tiedown fitting) is applied as described and divided by the g. load factor we find the fitting installation will be capable of restraining a 397-lb load under emergency conditions.

(6) When a cargo-restraining net or cargo container with multiple attachments is used, the static load requirements for each tiedown fitting may be divided equally between the fittings. For example, assume that the maximum cargo load to be carried is 1,800 lbs and 10 tiedown fittings are to be used, the static load requirement for each fitting is approximately 2,155 lbs.

Example: static load for each tiedown fitting

$$9g \times 1.33 \times 1{,}800 = 2154.6$$

Placard individual tiedowns for the maximum weight to be secured.

1204. OPERATING LIMITATIONS, LOADING INSTRUCTIONS, AND PLACARDS.

a. General. Revisions or supplements to the approved portions of the aircraft's Flight Manual markings, placards, or other operating limitations require FAA engineering approval. Submit the requested changes and supporting data to the local FAA Flight Standards District Office for review and processing.

b. Operating Limitations and Loading Instructions.

(1) Prepare revisions or supplements to the aircraft's Flight Manual or operating limitations, weight and balance records, and equipment list changes as necessitated by the installation of the cargo tiedown systems.

(2) Provide instructions covering the installation and use of the cargo restraint system. For aircraft that require a Flight Manual, incorporate these instructions as a supplement. On other aircraft, provide a placard that references the appropriate instruction. In the instructions, cover such items as removal and reinstallation of seats or other equipment exchanged for cargo restraint systems, use of cargo nets, barrier nets, number and positioning of tiedown straps, maximum load for each compartment or tiedown area, permissible load per square foot, number of tiedown points allowable per foot of track, and maximum height of the load's c.g. above the floor.

c. Placards: Cargo Area Placards. Install placards or other permanent markings to indicate the maximum allowable cargo load and weight per square foot limitation for each cargo area. Placard seat tracks as to number of tied own points permissible per foot of track. Attach a permanent label or other marking on each cargo net, barrier net, and at cargo tiedowns to indicate the maximum cargo weight that the net or attachment will restrain when installed according to the loading instructions.

1205. AIRWORTHINESS COMPLIANCE CHECK SHEET: CARGO TIEDOWN DEVICE INSTALLATIONS.

a. General. Cargo tiedown devices installations that are the same as those made by the manufacturer, or other installations which are already approved, may be accepted without further investigation. On other installations, the following points should be checked to determine that the installation is satisfactory.

b. Applicable Civil Aviation Regulations.

(1) 3.390 - Seats and Berths.

(2) 6.355 – Seats and Berths.

c. Applicable Title 14 of the Code of Federal Aviation Regulations.

(1) Part 21, § 21.303 – Replacement and Modification Parts.

(2) Part 23, § 23.785 – Seats, Berths, Litters, Safety Belts, and Shoulder Harnesses.

(3) Part 23, § 23.787 – Baggage and Cargo Compartments.

(4) Part 27, § 27.785 – Seats, Berths, Litters, Safety Belts, and Shoulder Harnesses.

(5) Part 27, § 27.787 – Cargo and Baggage Compartments.

(6) Part 29, § 29.785 - Seats, Berths, Litters, Safety Belts, and Shoulder Harnesses.

(7) Part 29, § 29.787 - Cargo and Baggage Compartments.

d. Structural Requirements.

(1) If changes or alteration of the aircraft structure are made, is the original strength and integrity of the structure retained? (Ref. AC 43.13-2B, chapter 1 and § 23.561.)

(2) Has the extent the modification affects the c.g. of the aircraft been evaluated? (Section 23.1589.)

(3) If the equipment is mounted either on the existing structure or on a bracket attached to the existing structure, is all of the structure (including the bracket, if used) adequate to support the required loads? (Ref. §§ 23.307, 23.613, 23.561, 25.307, 25.613, 25.561, 27.307, 27.613, 27.561, 29.307, 29.613, and 29.561.)

(4) Is the equipment installed so that it does not adversely affect other structures (either primary or secondary)? (Ref. § 23.1431.)

(5) Are means provided to permit proper inspections of the installation and related adjacent parts as components? (Ref. § 25.611.)

e. Hazards to the Aircraft or its Occupants.

(1) Does the modification create any projections that may cause injury by human impact?

(2) Does the fabric used in the modification comply with flame-resistant requirements?

(3) Does the modification affect the accessibility of the exits and doors adversely affected?

f. Detail Design Standards.

(1) Are suitable materials used in the construction, including standard fasteners, and will the method of fabrication result in a consistently sound structure? (Ref. §§ 21.305, 23.603, 23.605, 23.607, 23.613, 25.603, 25.605, 25.607, 25.613, 27.603, 27.605, 27.607, 27.613, 29.603, 29.605, 29.607, and 29.613.)

(2) Do cargo tiedown devices conform to an acceptable standard?

g. Instructions for Continued Airworthiness.

(1) Are there written procedures concerning equipment installation and removal procedures?

(2) Are there written equipment serviceability requirements?

(3) Are placards installed? (Ref. Paragraphs 9 through 12.)

(4) Are revisions or supplements provided for the aircraft's Flight Manual or operating limitations, if required? (Ref. Paragraphs 9 through 12?)

(5) Are there written scheduled inspection requirements to ensure the aircraft structure, tiedown devices, nets, and fittings are in serviceable condition?

(6) Are drawings available? (Ref. Figure 12-6.)

h. Recordkeeping.

(1) Has a maintenance record entry been made? (Ref. § 43.9.)

(2) Has the equipment list and weight and balance been revised? (Ref. Order 8310.6 chapter 1.)

(3) Has FAA Form 337 and Instructions for Continued Airworthiness been completed and accepted by the FAA?

1206. THRU 1299. RESERVED

FIGURE. 12-6. TYPICAL ALLOWABLE CARGO LOADING DIAGRAM

AERODESIGN Aircraft Engineering	report 333-1	sheet E3
16006 Canard Circle	prepared by	R.Howard
Austin, Texas 78734	checked by	R. Howard
(512) 266-0610 fax (512) 266-0609	revision	J