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DC-9 SAFE BOMB LOCATION STUDY

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for

National Aviation Facilities Experimental Center

Atlantic City, New Jersey 08405

JANUARY 1973

FINAL REPORT

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16. Abstract

The DC-9 commercial aircraft structure and systems are reviewed to determine the optimum location to place a discovered explosive device for the best chance of survival of the aircraft and minimization of casualties. The damage that would be caused by the explosion and the structural residual strength are determined.

The study revealed that the best place to put the bomb to do the least damage would be on the ventral stairs for those airplanes so equipped. On other DC-9 airplanes, the bomb should be put in the tail cone.

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PREFACE

This report determines the safe bomb location for the DC-9 commercial airplane. The study was performed by Douglas Aircraft Company for the Federal Aviation Administration, NAFEC, Atlantic City, New Jersey. Mr. Joseph J. Jaglowski, Jr. was the project manager.

This investigation was initiated following a rash of airplane hijackings and bomb extortion threats. The FAA has initiated a many-pronged attack on the general problem; this being only one of a number of possible solutions. It is hoped that this effort will in some small way contribute to reinforcing public confidence.

The author wishes to express his appreciation to Mr. Robert G. S. Sewell of Naval Weapons Center, China Lake, California for his valuable assistance.

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1 INTRODUCTION

This study, sponsored by the FAA, is to determine an optimum location within a DC-9 commercial aircraft to place a discovered explosive device for the best chance of survival of the aircraft and minimization of casualties.

An analysis is performed for hole size and airplane strength of the residual structure around the hole caused by the detonation of an explosive charge. The explosive charge is assumed to be equivalent to 4 pounds (8 sticks) of 40 percent nitroglycerine based commercial dynamite.

The specific tasks are:

Task 1 - Study for optimum bomb location

- Identify critical control systems in the fuselage.
- Identify critical piping in the fuselage.
- Identify critical electrical wire in the fuselage.
- Review stress/analysis for areas of maximum margin of safety consistent with minimum systems locations.

Task 2 - Analysis

- Review fuselage structure for minimum margins of safety.
- Perform analysis for hole size.
- Perform analysis for aircraft strength.

Section 2 provides a general description of the major subsystems of the DC-9 airplane. Section 3 discusses the best location to place the bomb, identifies the critical equipment which would be affected and justifies the selection of three locations to be analyzed. The three locations are: in the forward fuselage hat rack, in the tail cone and on the opened ventral stairs. Section 4 analyzes the effect of detonating the bomb in the forward hat rack. In many airplanes, the bomb must remain in the cabin and this location represents the typical option available. It is indicative of the damage done and the residual strength remaining if a bomb of this size were to be detonated in many of today's modern jet aircraft. The basic blast effects are discussed and the shock wave parameters for this size bomb are derived. The blast damage criteria is defined. The damage which would occur and the strength that would remain are calculated. Section 5 determines the damage done if the bomb were detonated in the tail cone or on the ventral stairs. This airplane is fortunate to possess an area out of the main cabin which can be entered in flight. A large portion of the tail section is non-critical and makes an ideal candidate for the safe bomb location. Even better, most DC-9 airplanes have ventral stairs on which a bomb can be placed and lowered outside the contour. Section 6 describes three previous incidents where airplanes have sustained massive damage and still landed safely. Section 7 presents the conclusions.

2 AIRPLANE DESCRIPTION AND OPERATION

This section describes the airplane and some of its major systems (see Figure 2-1). The airplane structure features an all metal, fully cantilevered sweptback wing, semi-monocoque constructed fuselage, and the empennage or tail section. The structure incorporates aluminum alloy external sheets stiffened with rolled and extruded hat-section longerons which are attached to rolled or extruded Z-section frames, or machined channels. A three-view drawing of a typical configuration is shown in Figure 2-2.

FUSELAGE STRUCTURE

The fuselage is an all-metal semi-monocoque structure. The fuselage consists of a nose section, center section, and a tail section. The nose section contains the flight compartment, nose gear well, electrical/ electronics compartment, and forward accessory compartment. The center section contains the passenger compartment and the forward and aft lower cargo compartments. The tail section contains the aft accessory compartment and the APU compartment. Figure 2-3 and 2-4 show station diagrams for the full range of airplanes.

The main frame is constructed of transverse frames, longitudinal stiffeners, and lateral floor beams. Except for the forward part of the nose section and the aft part of the tail section, the fuselage consists of two semicircular joined segments. The upper segment has a radius of approximately 66 inches and the lower segment a radius of approximately 62 inches.

Transverse frames are located approximately 19 inches apart throughout the fuselage. The frames are made in two sections. The sections are joined in the passenger and flight compartments by the floor beams, forming a slight cusp. The upper and lower frame sections aft of the passenger and service door openings are constructed of Z-shaped rolled sheet aluminum alloy, except the frame sections in the passenger compartment over the center wing. Frames over the center wing area are made of C-shaped extruded aluminum alloy above the window belt, and forgings and machined sections below the window belt.

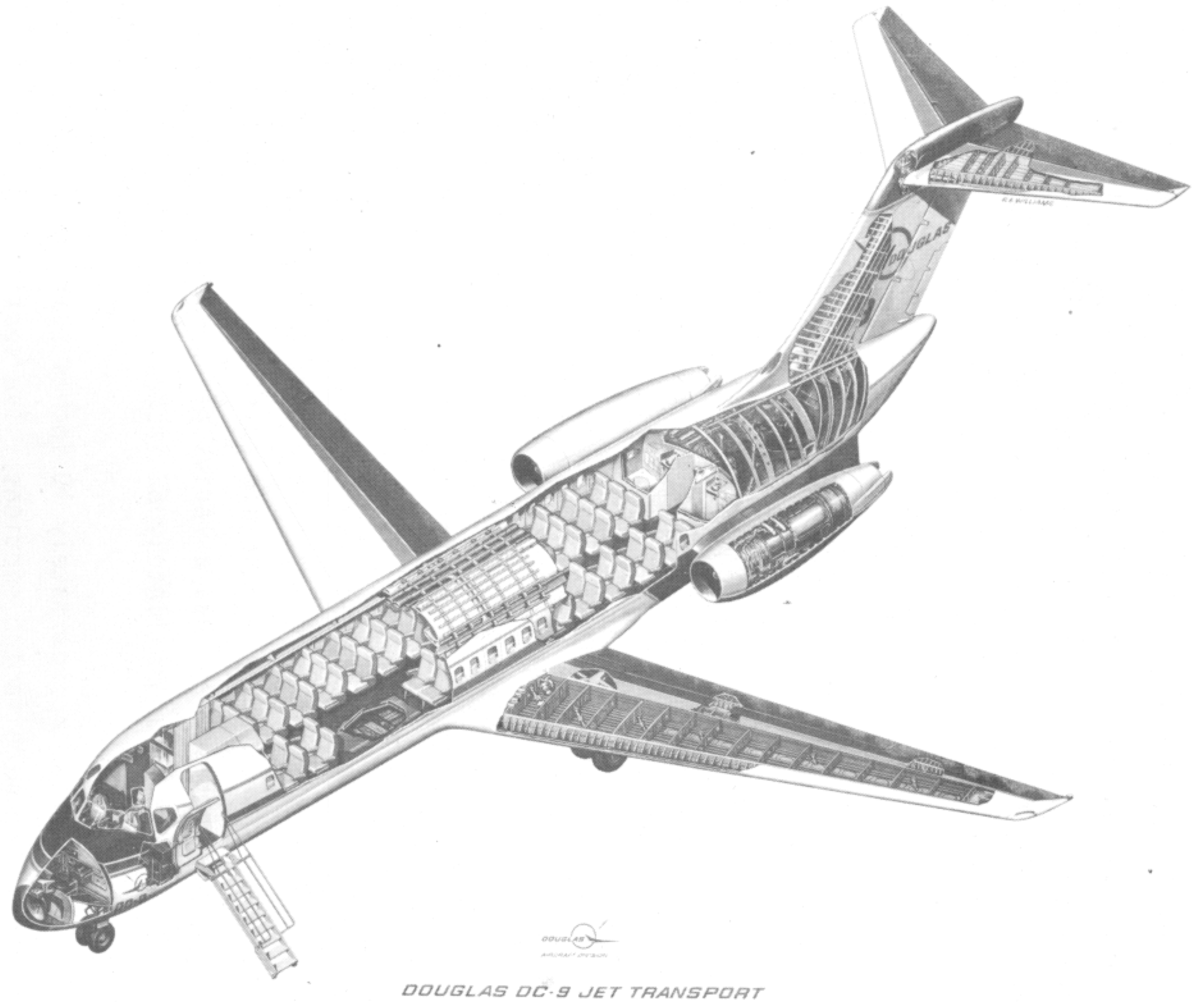


FIGURE 2-1. CUTAWAY VIEW OF DC-9

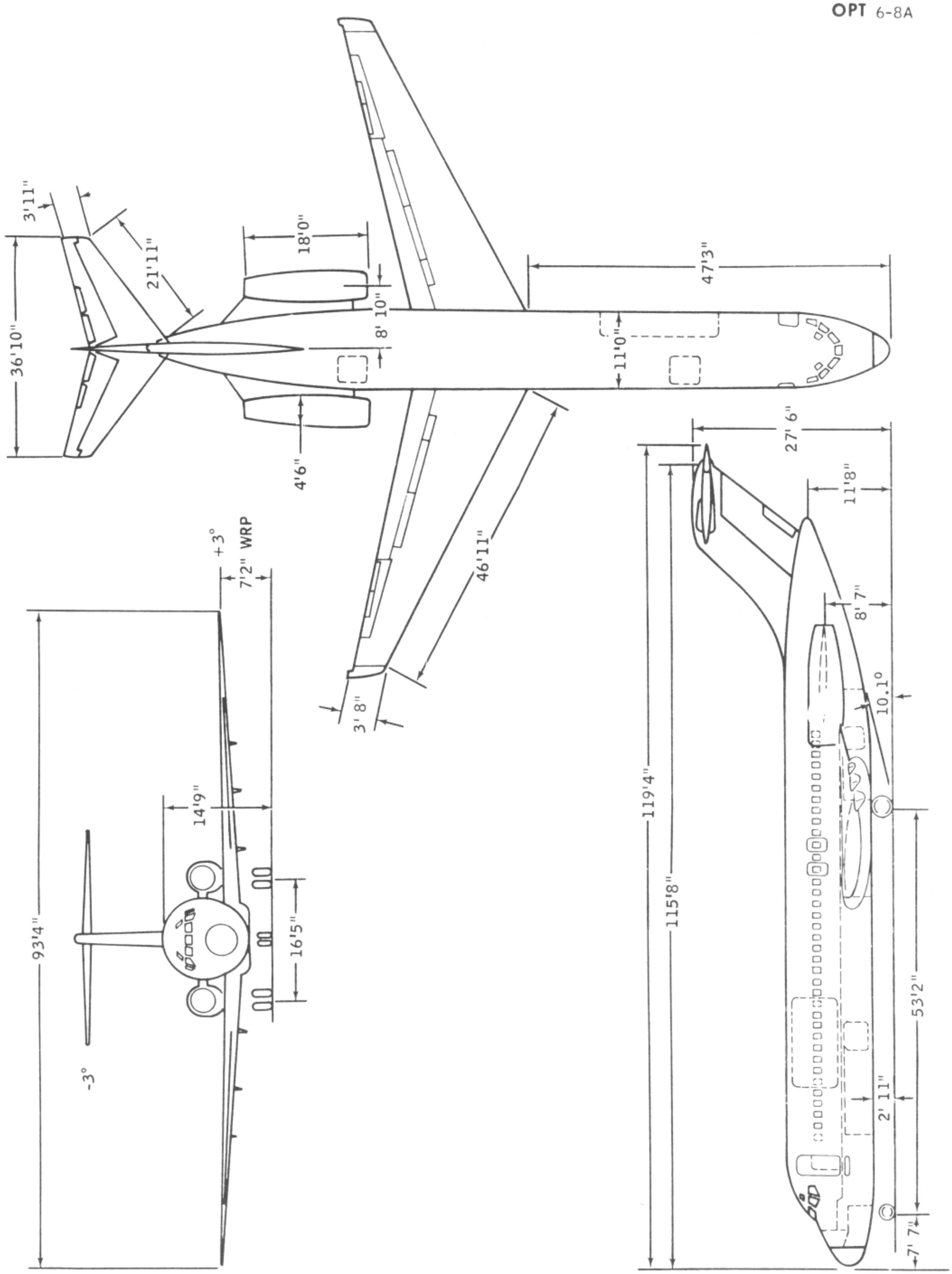


FIGURE 2-2. DC-9 SERIES "30" THREE VIEW

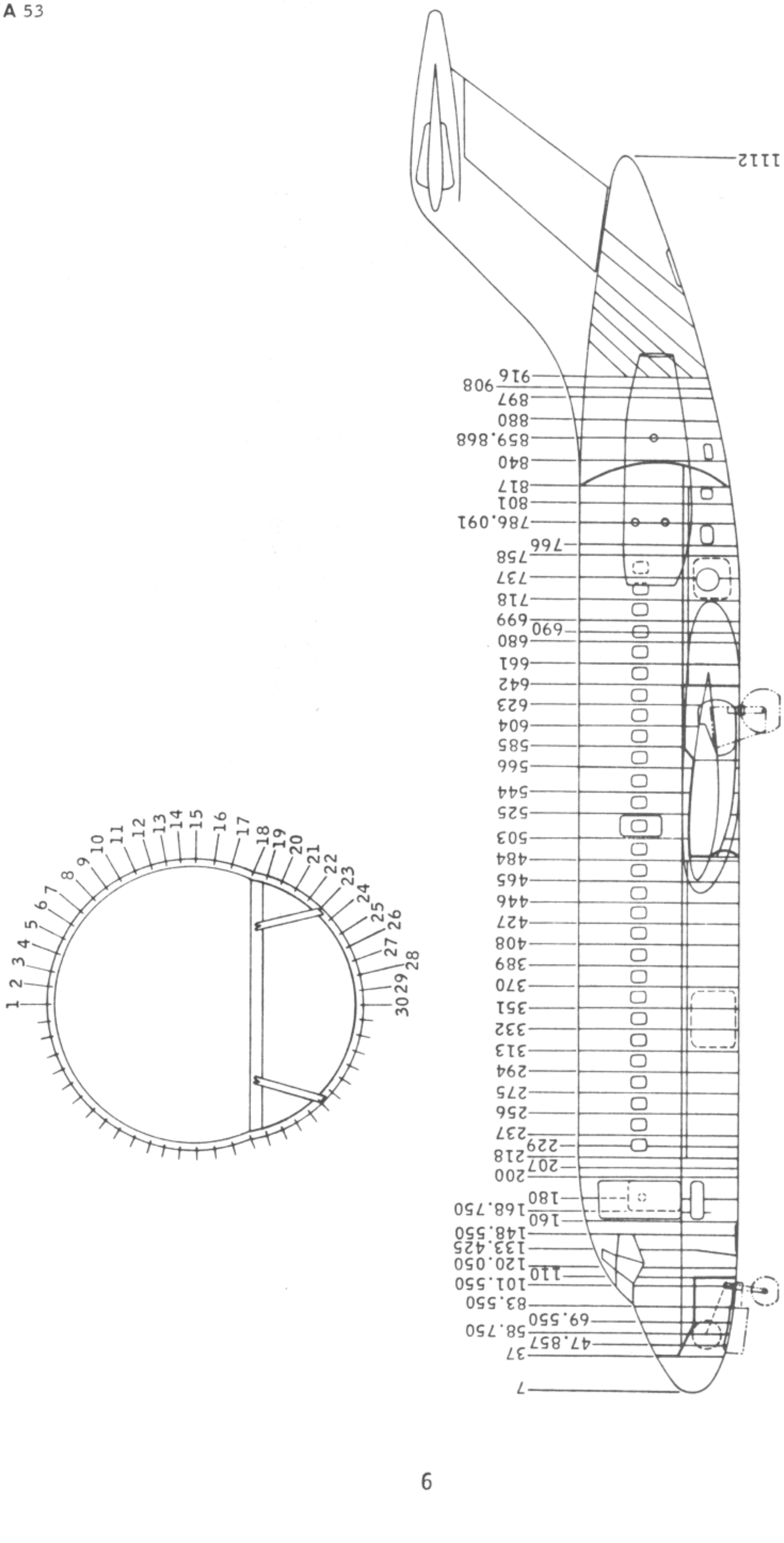


FIGURE 2-3. FUSELAGE STATION DIAGRAM — SERIES 10

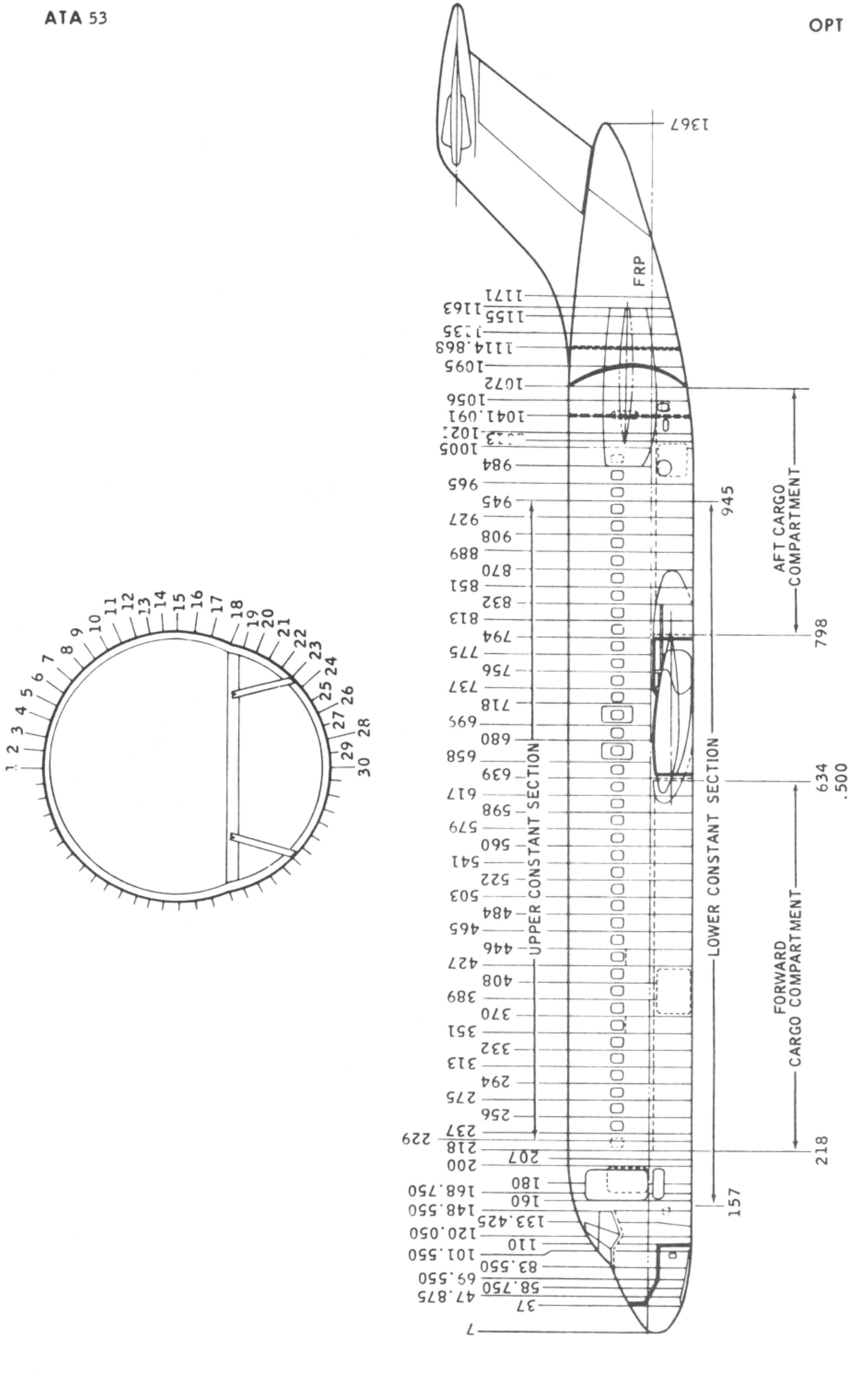


FIGURE 2-4. FUSELAGE STATION DIAGRAM — SERIES 40

Engine support frames are installed forward and aft of the pressure bulkhead, located at the aft end of the passenger compartment, to distribute engine loads over a large area. The frames are a built-up type, consisting of doublers, angles, channels, fittings, and webs. Frames in the flight compartment are larger than frames in the rest of the fuselage and are made of heavier material to provide additional structural strength. Frames around all door and window openings in the flight compartment are made of heavier material than adjacent frames and are reinforced with doublers, intercostals, and fittings. Formed pans are installed around openings of windows in the passenger compartment for installation of window panes and to strengthen the plating.

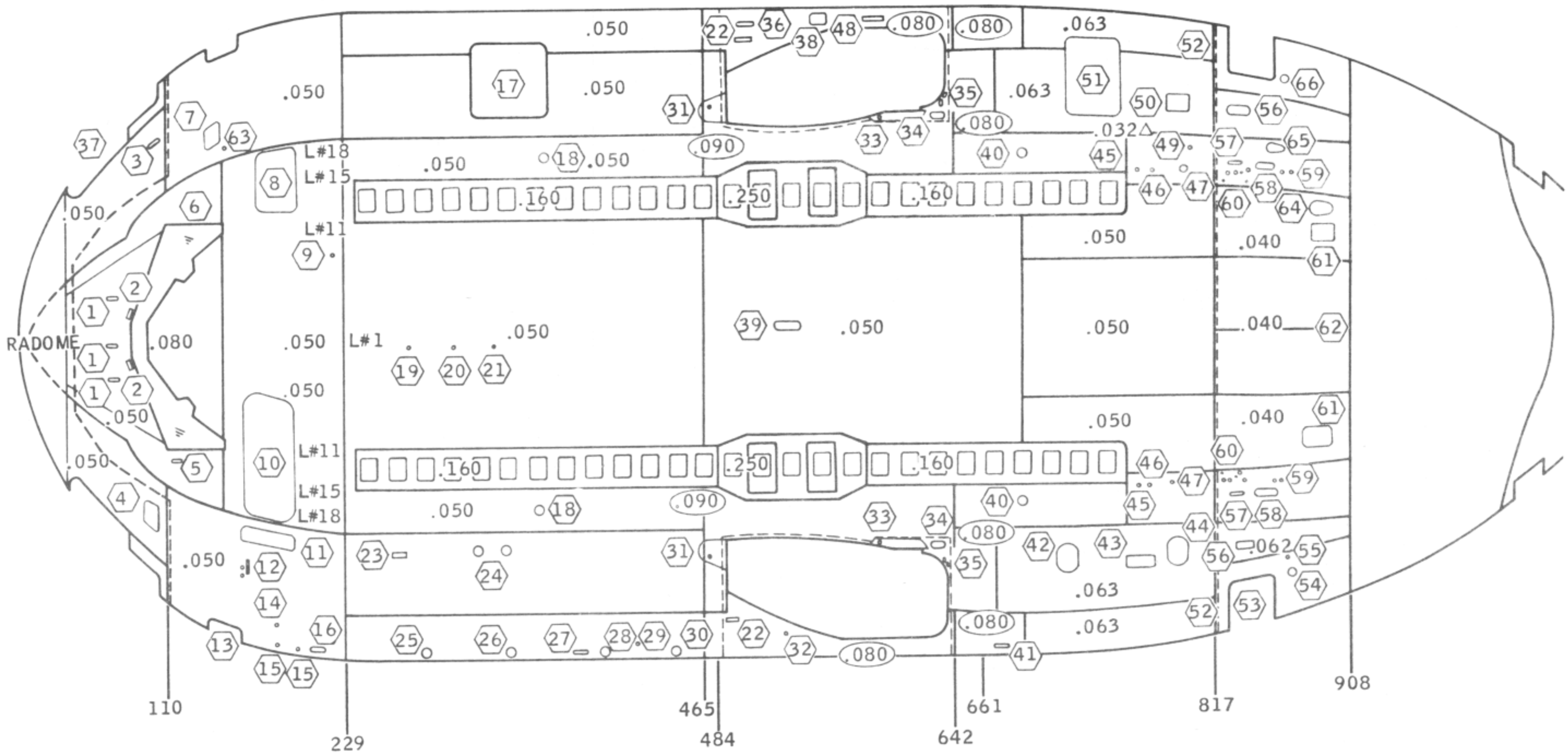
Longerons are located around the perimeter of the fuselage at intervals of approximately 7-1/2 inches. The longerons are attached to the frames and provide the main longitudinal attachment for the exterior plating. The longeron sections are joined together by fittings.

The fuselage exterior covering consists of skins of aluminum alloy and titanium. (See Figure 2-5.) The skins are attached to longerons, formers and doublers to form panels. The panels are attached to the main frames of the fuselage by permanent fasteners. The splice doublers and longeron splice fittings are used to join the panels vertically. The panels are joined horizontally by doublers, longerons, and intercostals. Typical construction is shown in Figure 2-6.

The panels containing window openings for the passenger compartment are milled along the upper and lower edges to provide a flush lap joint. Intercostals are installed along the flush lap joints between the transverse frames. All panel joints on exposed sections of the fuselage, forward of the pressure dome, are flush joints except lap joints along the top centerline of the fuselage.

Titanium panels are used on the sides of the fuselage adjacent to the engines and in the lower surface of the fuselage in the area of the APU unit. The titanium skins protect the fuselage from excessive heat conducted by the engine and APU unit and the engine and APU exhaust gases.

Lateral beams and longitudinal intercostals support the floor of the flight and passenger compartments. The beams in the flight compartment and at the



CORRESPONDING STATION VALUES		
SERIES -10 SERIES -20	SERIES -30	SERIES -40
642 661 817 908	756 775 996 1087	794 851 1072 1163

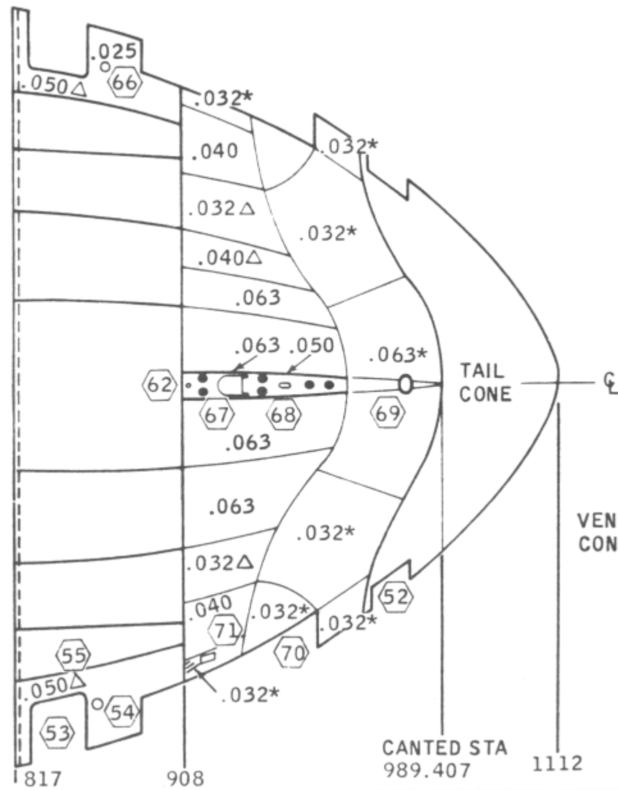
NOTE:

- - - - INDICATES PRESSURE BOUNDARY

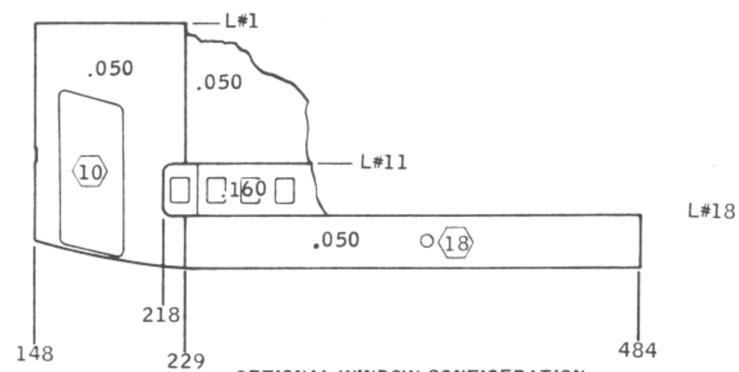
SKIN GAGES WITH
* ARE 2024
(oval) ARE 7075
△ ARE TITANIUM
& ALL OTHERS ARE 2014

• INDICATES LIGHTENING HOLE CUTOUT

FIGURE 2-5. FUSELAGE PLATING DIAGRAM (SHEET 1)



VENTRAL STAIRWAY CONFIGURATION



OPTIONAL WINDOW CONFIGERATION LEFT SIDE ONLY

CORRESPONDING STATION VALUES

SERIES-10 SERIES-20	SERIES-30	SERIES-40
817	996	1072
908	1087	1163
989.407	1168.407	1244.407
1112	1291	1367

⬡ PLATING CUTOUTS

1. PITOT TUBE
2. WINDSHIELD WIPER DOORS
3. AIR TEMPERATURE INDICATOR
4. ELECTRICAL SERVICE
5. ENTRANCE DOOR - HOLD OPEN
6. GALLEY DOOR - HOLD OPEN
7. WATER SERVICE
8. SERVICE DOOR
9. GALLEY VENT
10. ENTRANCE DOOR
11. AIRSTAIR DOOR
12. HANDLE & POWER SWITCH FOR AIRSTAIR DOOR
13. ACCESS DOOR (RADIO COMPARTMENT)
14. STAIRWAY SUMP DRAIN
15. MARKER BEACON ANTENNA
16. ATC ANTENNA
17. FWD CARGO DOOR
18. LIGHT INSTL., WING SCANNING
19. ADF LOOP ANTENNA
20. ADF LOOP ANTENNA
21. VHF ANTENNA
22. VENT - UNDERWING COMPARTMENT
23. RADIO RACK EXHAUST VENT
24. PRESSURE RELIEF VALVES
25. RADIO ATM ANTENNA
26. RADIO ATM ANTENNA
27. DME ANTENNA
28. RADIO ATM ANTENNA
29. VHF ANTENNA
30. RADIO ATM ANTENNA
31. PNEUMATIC DEICING SYSTEM
32. VHF ANTENNA
33. HYDRAULIC LINES
34. FUEL LINE
35. FLAP BUS CABLE
36. DME ANTENNA
37. NOSE WHEEL DOOR
38. FUEL TANK VENT
39. ANTI-COLLISION LIGHT
40. LIGHT INSTL., ENGINE NACELLE SCANNING
41. ANTI-COLLISION LIGHT
42. CABIN AIR OUTFLOW BUTTERFLY VALVE
43. CABIN AIR OUTFLOW NOZZLE VALVE
44. TOILET SERVICE
45. ENGINE CONTROL - CONTROLS FSO
46. ENGINE CONTROL - THROTTLE
47. FUEL LINE
48. LANDING GEAR DOOR SAFETY LATCH
49. AFT LAVATORY VENT
50. WATER SERVICE
51. AFT CARGO DOOR
52. AFT FUSELAGE ACCESS DOOR
53. A.P.U. ACCESS DOOR
54. GROUND PNEUMATIC CONNECTION
55. FUEL SHROUD LINE DRAIN
56. THRUST REVERSE VALVE ACCESS DOOR
57. MAIN PNEUMATIC LINE
58. ELECTRICAL CONDUITS
59. HYDRAULIC LINES
60. HYDRAULIC LINES
61. AIR CONDITIONING DUCT EXHAUST
62. DEICING DUCT
63. OXYGEN DRAIN
64. A.P.U. EXHAUST
65. A.P.U. COMPARTMENT COOLING EXHAUST
66. AIR CONDITIONING GROUND CONNECTION
67. AIR CONDITIONING DUCT
68. DEICING DUCT
69. RUDDER TORQUE TUBE
70. VENTRAL STAIRWAY DOOR
71. GROUND ATTENDANTS CONTROL FOR DOOR

FIGURE 2-5. FUSELAGE PLATING DIAGRAM (SHEET 2)

FIGURE 2-6. TYPICAL FUSELAGE CONSTRUCTION

passenger door are a built-up type, consisting of plating, extruded and rolled sheet angles and channels, doublers, gussets, and fittings. The beams in the passenger compartment are extruded sections. The upper and lower sections of the transverse frames are attached to the outboard ends of the beams. Braces are attached to the lower edge of the beams, approximately 24 inches inboard of each end. The braces extend downward diagonally to the lower section of the transverse frames and support the beams. When the cargo compartment lining is attached to the braces, a tunnel is formed along each side of the cargo compartments for cables, wiring, tubing, and equipment.

A bulkhead is installed at the forward end of the fuselage. Bulkheads are also installed between the flight and forward accessory compartments, nose gear well and electrical/electronics compartment, forward accessory compartment and nose gear well (canted), electrical/electronics and forward lower cargo compartments, lower cargo compartments and main gear wells, and passenger and aft cargo compartments and tail section. All bulkheads are constructed of webs reinforced by extruded and rolled sheet angles and channels, doublers, gussets, and fittings.

The auxiliary structure consists of the floor of the passenger and flight compartments, aft accessory compartment walkway, aft lavatory partitions, electrical/electronics compartment access step, and APU compartment enclosure.

The floor consists of panels of various lengths and widths supported by beams, intercostals, and seat tracks. (See Figure 2-7.) Floor panels in the forward section of the flight compartment are made of fiberglass. Floor panels in the aft section of the flight compartment and the passenger compartment are made of aluminum alloy reinforced with hat sections of aluminum or fiberglass, depending on the thermal requirements of individual panels. Plywood is used in place of hat sections in the raised portion of the flight compartment floor. Insulation is attached to the underwide of the panels, except the panels installed in the flight compartment, passenger compartment outboard of the outboard seat tracks, and over the center wing.

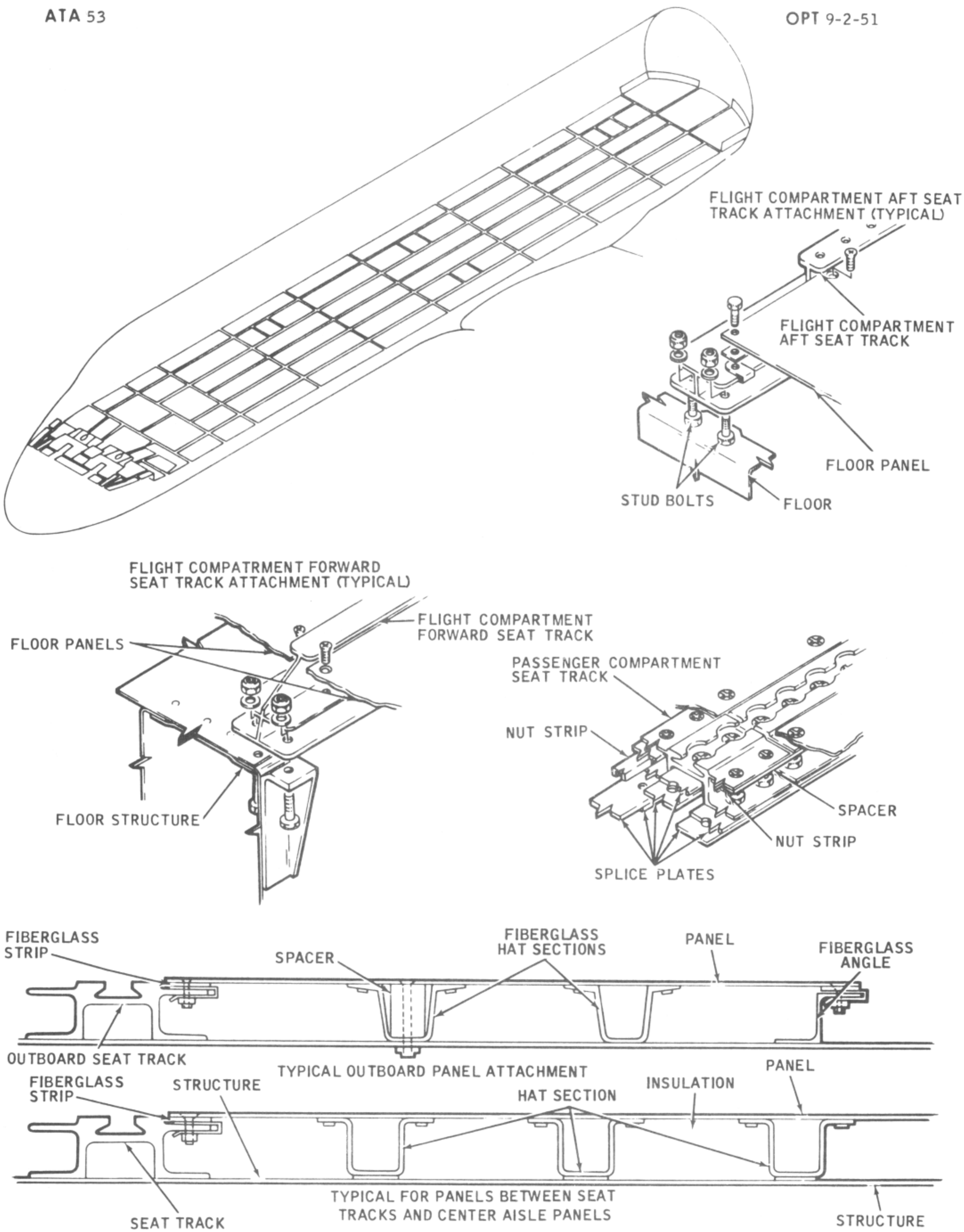


FIGURE 2-7. FLOOR STRUCTURE PANELS

The aft accessory compartment walkway extends from the aft side of the aft pressure bulkhead to the forward edge of the tail cone. The forward section of the walkway is level with the passenger compartment floor, while the aft section is canted upward to the tail cone. The walkway is constructed of corrugated aluminum alloy panels and stiffeners. Pads of safety walk are cemented to the walkway to provide a nonskid walking surface. The walkway provides access to components in the aft accessory compartment, and is used as an emergency exit from the passenger compartment, when the tail cone is jettisoned.

Some configurations have a ventral stairs. The stairway door hinges at the forward edge and contains steps which become the stairway in the open position and form the bottom of the fuselage in the closed position (see Figure 2-8). It is hydraulically operated and controlled by a lever just aft of the rear pressure bulkhead. Handrails are provided. The stairway tunnel side walls are fiberglass and the compartment walkway raises up to form the roof of the tunnel.

Two partitions, a header and a door enclose each aft lavatory. The partitions consist of outer panels of plywood, a honeycomb core, and hardwood blocking. The forward partition is installed on the forward side of the engine stiffening frame located forward of the aft pressure bulkhead. The fore-and-aft partition extends forward from the pressure bulkhead to the door. The door is located between the partitions and is attached to the fore-and-aft partition by a piano-type hinge. A header is located above the door and above the passageway.

The access step is located in the electrical/electronics compartment on the left side. The step is attached to the side of the fuselage directly below the access door in the floor of the flight compartment. A corrugated plastic sheet is attached to the fuselage structure in the area below the flight compartment floor supporting structure and the top of the step, to prevent damage to wiring that is attached to the sidewall of the compartment.

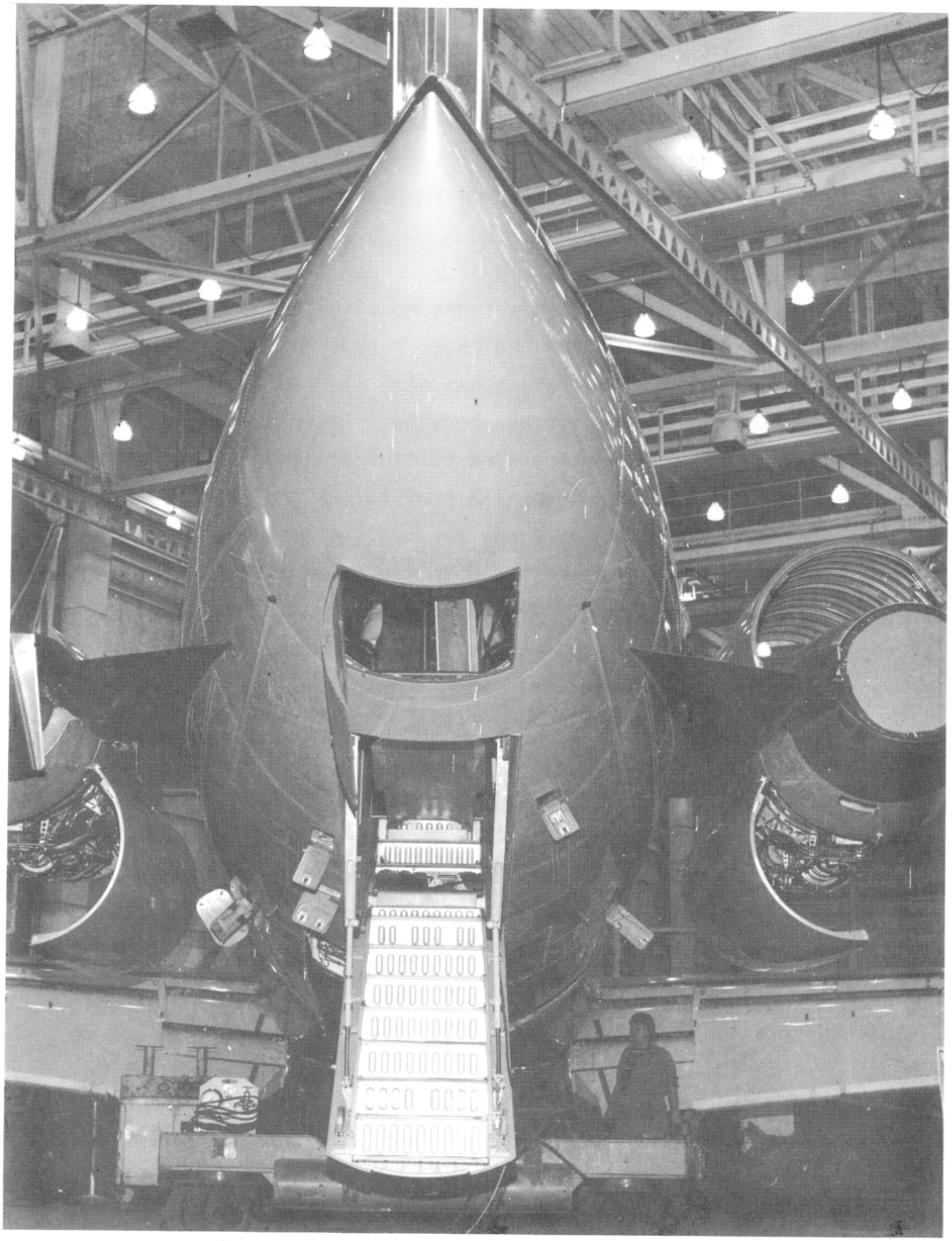


FIGURE 2-8. REAR VIEW SHOWING VENTRAL STAIRS AND CONE

The APU compartment enclosure is located in the fuselage lower structure aft of the pressure bulkhead. The enclosure forms a fireproof compartment for the APU and is covered with an insulation blanket. Only the blanket and small access panels are removable. The enclosure side and top panels are permanently attached to the fuselage structure and are not removable. The top and side panels are made of .020 titanium.

The nose radome is a streamlined antenna housing that protects the antennas attached to the forward fuselage bulkhead form structural damage and adverse environment conditions. The radome has high electromagnetic transmission characteristics, and low resistance to wind. The radome is constructed of glass fiber laminated skins separated by flutes which form a hollow core. A replaceable rain erosion protection boot covers the frontal area of the radome. The exterior surface of the radome is protected by weather-resistant paint. Lightning strips are attached to the outer surface of the radome to minimize damage by lightning. The radome is hinged at the top, and secured at the bottom by latches. Two supports hold the radome open. The supports are stowed on the forward bulkhead when not in use.

There are fuselage-to-wing fillets on each side of the fuselage to provide aerodynamic smoothness. The fillets are attached to the fuselage and fillet supporting structure with screws. A rubstrip is attached to the outer edges of the fillets to allow the wing to flex without causing material damage to the wings or fillets. Floodlight lenses are installed in the leading edge fillets and can be removed with the fillets without disturbing the floodlight assemblies. Individual fillet sections may be removed for access to the fuselage and wing area.

The tail cone is attached to the aft end of the fuselage and can be jettisoned. The tail cone is constructed of glass fiber laminated skins which are separated by flutes which form a hollow core. An access door is located in the lower forward section and provides access to the aft accessory compartment from outside the airplane without jettisoning the tail cone. Four spring-loaded latches, attached to the aft fuselage attach frame, engage four locks on the tail cone attach ring and secure the tail cone to the fuselage. A locking cable locks the latches in closed position (see Figure 2-9).

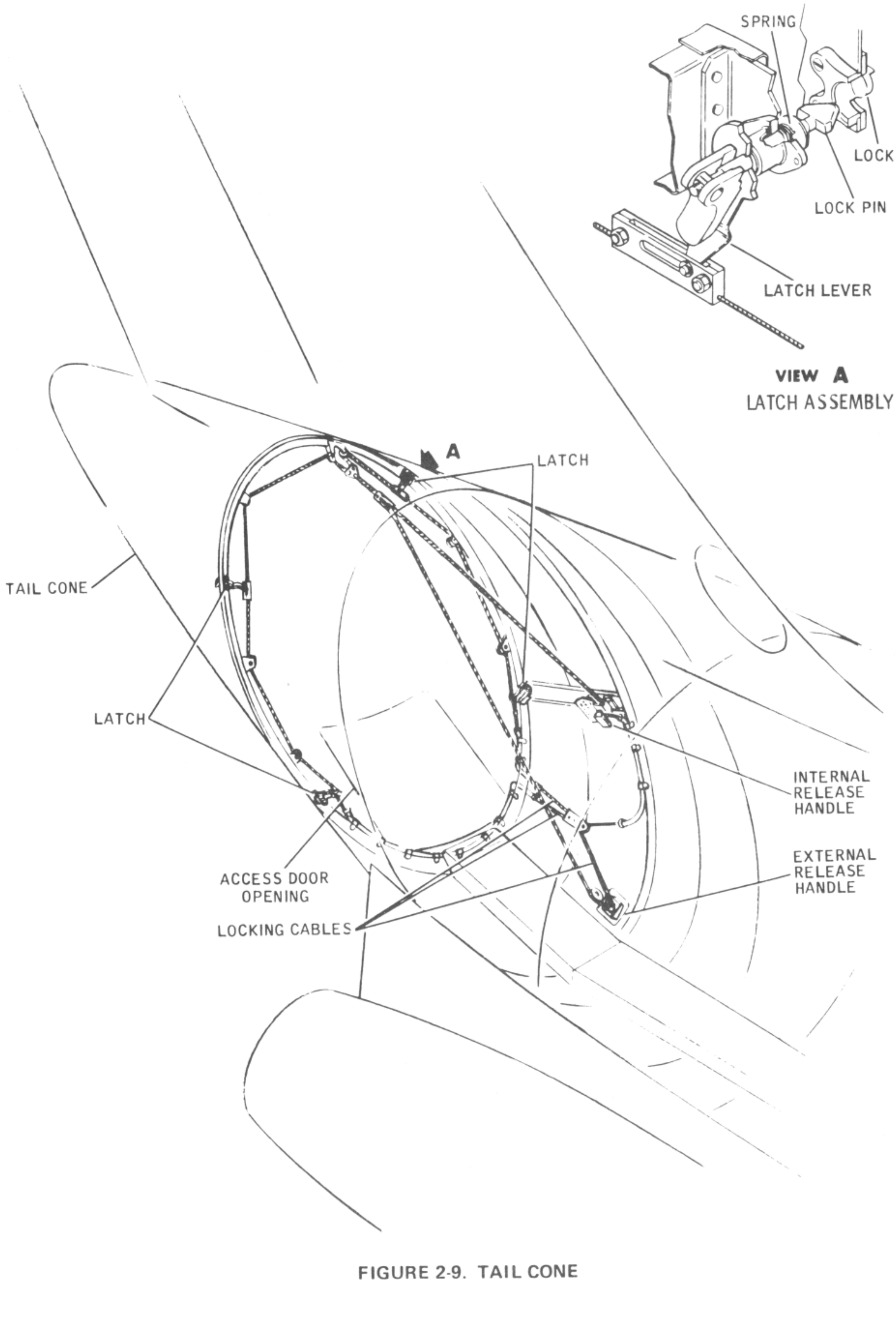


FIGURE 2-9. TAIL CONE

WING STRUCTURE

The wing is a sweptback, fully cantilevered, stressed all-metal skin, single-unit structure, mounted through the lower fuselage. The wing incorporates ailerons, spoilers, trailing edge flaps, integral fuel tanks, and supporting structure for the main landing gears (see Figure 2-10).

The wing is constructed with two spanwise spars, spanwise panel stiffeners, and chordwise ribs and bulkheads. The wing spars are constructed of extruded spar caps with sheet spar webs and extruded spar web stiffeners. At the wing tips the sheet metal web is eliminated by overlapping the two spar caps. The ribs between the spars utilize rolled and extruded stiffeners. The bulkheads that form the integral fuel tanks are attached and sealed at the spars and upper and lower wing skins. The wing skins are (Alclad) roll-tapered and butt-spliced, and are fabricated from 2024-T4 aluminum alloy on the lower surface, and 7075-T6 alloy on the upper surface. Stiffening elements are machined from extrusions. The skins, stringers, ribs, bulkheads, and spars form a cellular box capable of supporting the wing bending, shear, and torque loads. Flush riveting is used for all wing exterior surfaces.

The wing contains two integral fuel tanks between the front and rear spars. The center wing area is designed and sealed to permit incorporation of a center wing auxiliary fuel tank. All faying surfaces and corner fittings are precision fit to make an inherently fluid-tight tank, with sealing compounds considered secondary in the sealing of the tanks.

The local wing structure in the tank area forward of the landing gear attach points is designed for loads exceeding those of the basic gear. This design in load difference is incorporated to minimize the possibility of tank rupture in the event of gear failure or breakaway. Flow baffles installed in the main tank prevent the rapid shift of fuel spanwise, and form a gravity supplied fuel reserve at the inboard end of each main tank.

Stressed access doors are incorporated as a part of each integral fuel tank structure, to permit access to the system components within the tank.

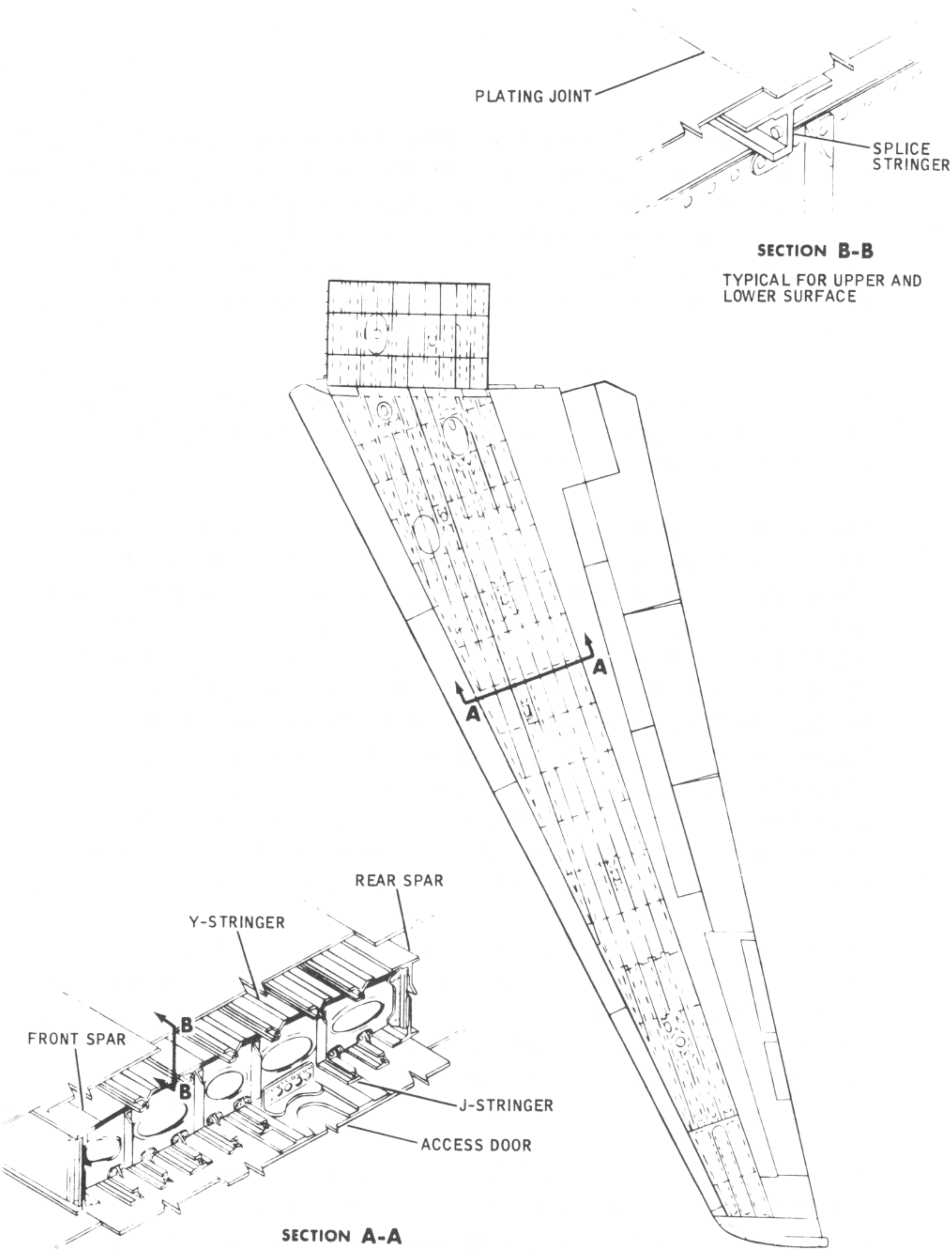


FIGURE 2-10. WING STRUCTURE

The leading edges of the wing are of double-skin construction and contain ducts for thermal anti-icing distribution. The leading edges attached to the front spar with screws and nut plates to permit repair or replacement. Wing leading edge sections are interchangeable between airplanes with replacement of wing leading edge splice belts. Access doors are provided in the leading edge section which permit access for servicing installations forward of the front spar.

Faired wing tips incorporate position and landing lights and are removable for repair or replacement. Electrical quick-disconnects are provided for position and landing light wiring.

Ailerons are mounted on truss-type brackets attached to the wing rear spar. The aileron hinges incorporate roller-type bearings which are replaceable without removing the aileron. The aileron is all metal construction, with a spar, ribs, and skins with bonded doublers. The ailerons are sealed to prevent entry of water. Drain holes are located in the lower surface. An all-metal control tab is incorporated in the trailing edge of each aileron. The tabs are sealed to prevent entry of water. The mass balance required for flutter prevention is integrally installed and the tabs are interchangeable without the necessity of rebalancing the aileron. Trim tabs are also incorporated in the trailing edge of each aileron just outboard of the control tab. The trim tab is sealed to prevent entry of water, balanced for flutter prevention, and is interchangeable without rebalancing the aileron.

Hydraulically operated flaps are installed in the trailing edge portion of the wing, inboard of the ailerons. Each flap is supported on one sideload track in the fuselage and three external hinges on the wing. The flaps are designed with replaceable wear strips where the flaps contact the seals, fuselage, or wing structure. Flaps are constructed with an internal framework consisting of spars, ribs, and skins with bonded doublers.

The entire aft section of each flap is removable at the spars. The aft section is composed of five parts which are attached to the sparcaps with screws and are joined at the ends by links to provide expansion joints. The flap leading edge is removable in sections for access, inspection, repair, or replacement.

The flap vanes are constructed of skins, ribs, and metal honeycomb, and are attached to the flaps with adjustable fittings for gap control. The trailing edges of the flaps are fabricated from fiberglass. The outboard and center spoilers provide lateral control in conjunction with the ailerons, and act as speedbrakes both in flight and on the ground. The spoilers are sealed to prevent entry of water.

EMPENNAGE STRUCTURE

The empennage, or tail group, consists of a vertical stabilizer, horizontal stabilizer, two elevators, and a rudder (Figures 2-11 through 2-14). The vertical stabilizer is attached to the aft fuselage and the horizontal stabilizer is mounted at the top of the vertical stabilizer. The rudder and elevator surfaces are mounted on the vertical and horizontal stabilizers, respectively. The vertical stabilizer is a fully cantilevered sweptback type, constructed as an integral part of the aft fuselage structure. The major portion of the vertical stabilizer leading edge is interchangeable with replacement of the leading edge splice doublers. A section of the leading edge is constructed of laminated fiberglass and contains the VOR antenna.

The vertical stabilizer is constructed with two spars joined by chordwise ribs and covered with aluminum alloy skin panels stiffened with spanwise stringers. The spars are a flat webbed-type with extruded or formed stiffeners and caps. The leading edge is bolted to the front spar. A section of fairing above the horizontal stabilizer is hinged for access to the horizontal stabilizer mechanism.

The horizontal stabilizer is of all-metal construction joined by chordwise ribs and covered by integrally stiffened machined panels between the spars. The spars are flat webbed-type having caps, extruded or formed stiffeners, and webs. Adequate access to parts requiring maintenance is provided.

The elevators are of all-metal construction with ribs and skins having bonded doublers. The elevators are sealed to prevent entry of water and excessive circulation of air, and are provided with drain openings. Elevators are

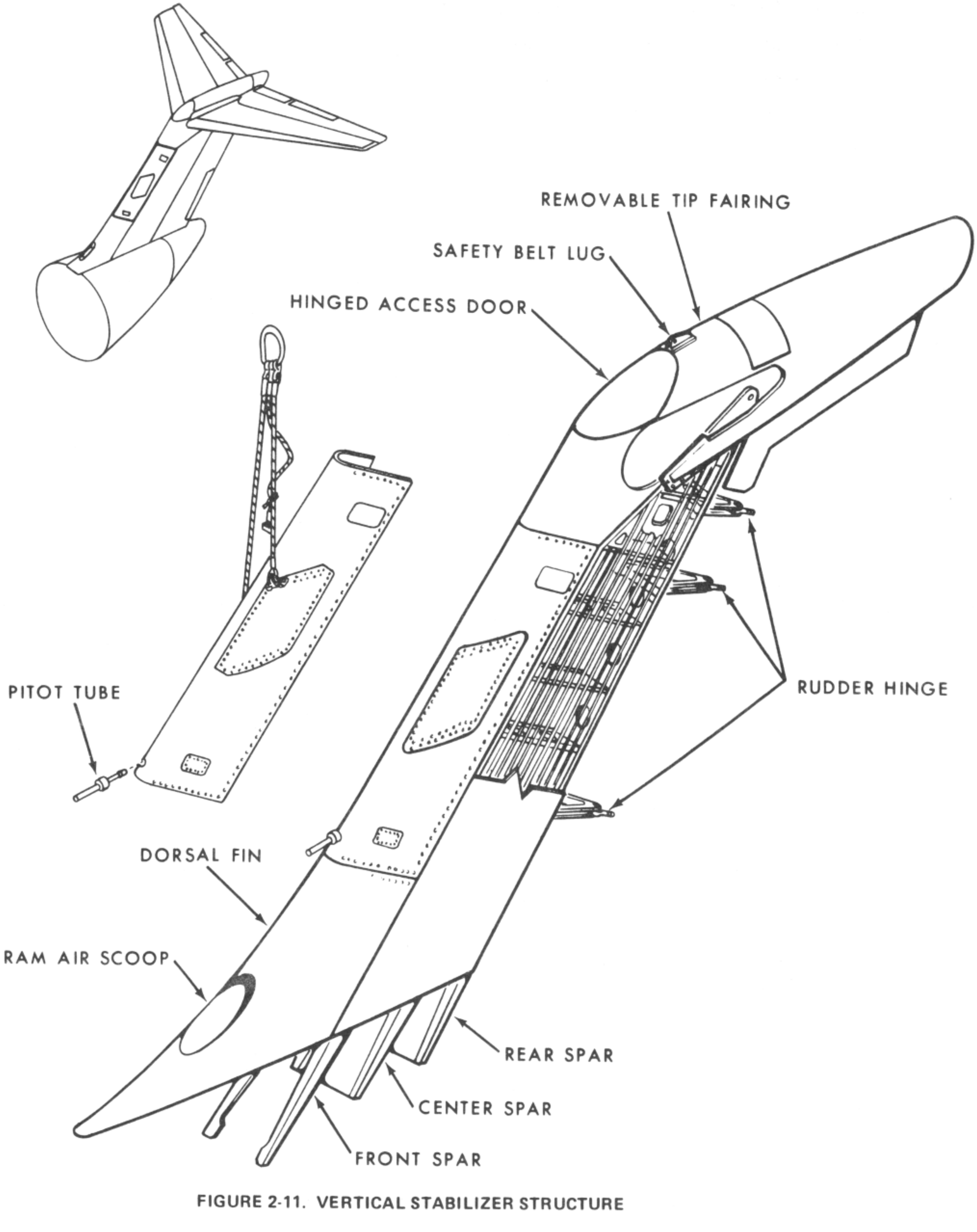


FIGURE 2-11. VERTICAL STABILIZER STRUCTURE

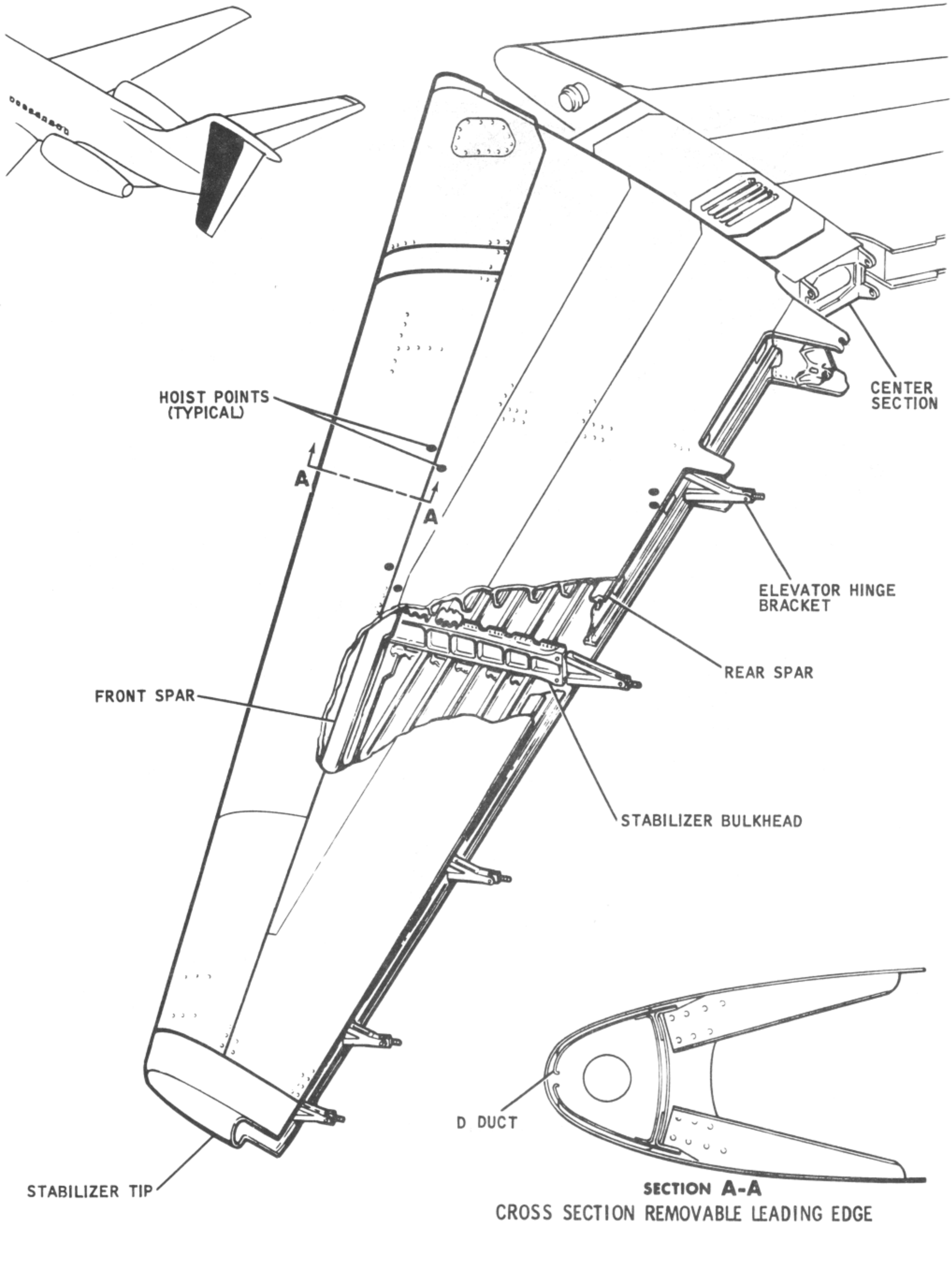


FIGURE 2-12. HORIZONTAL STABILIZER

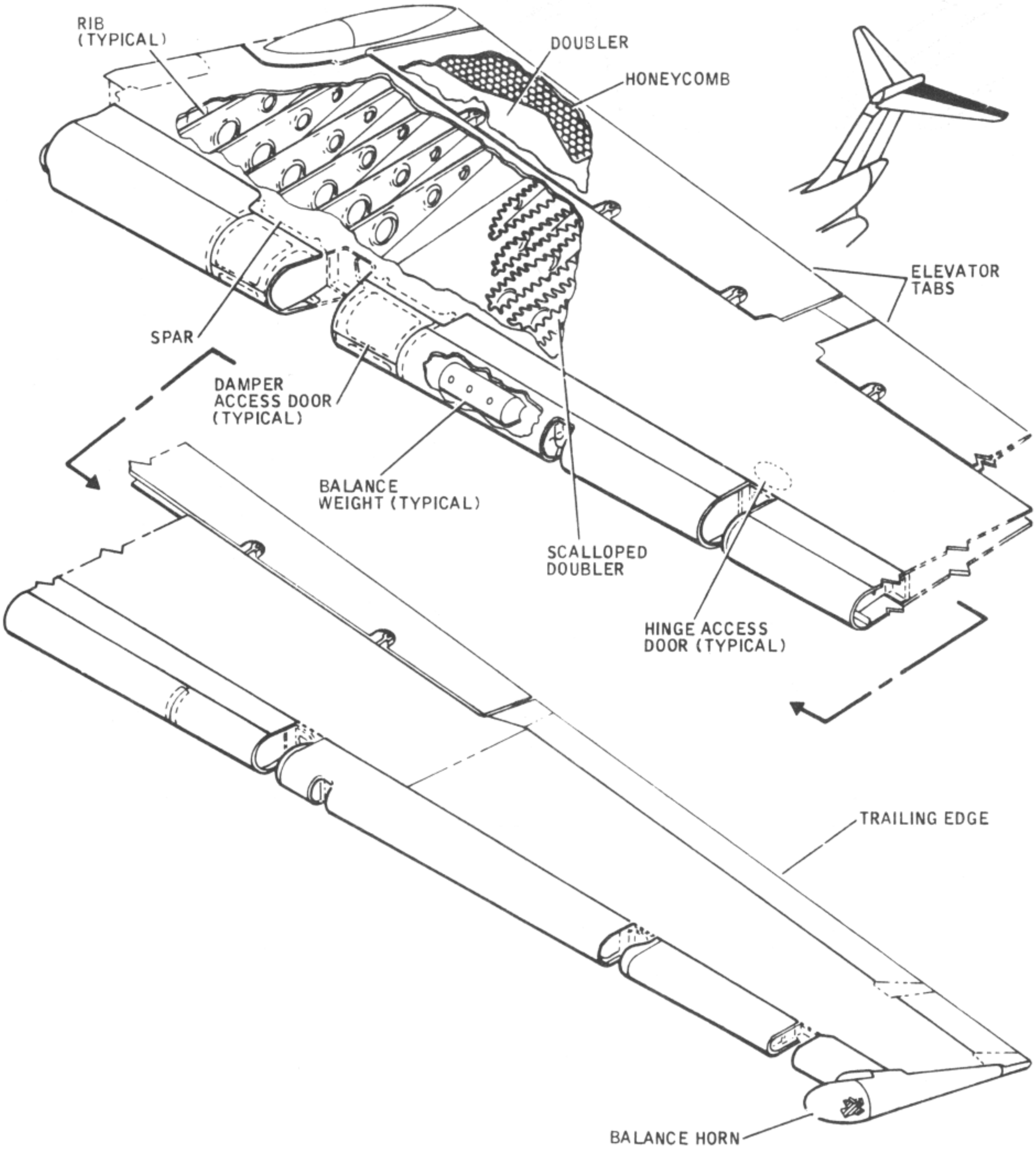


FIGURE 2-13. ELEVATOR

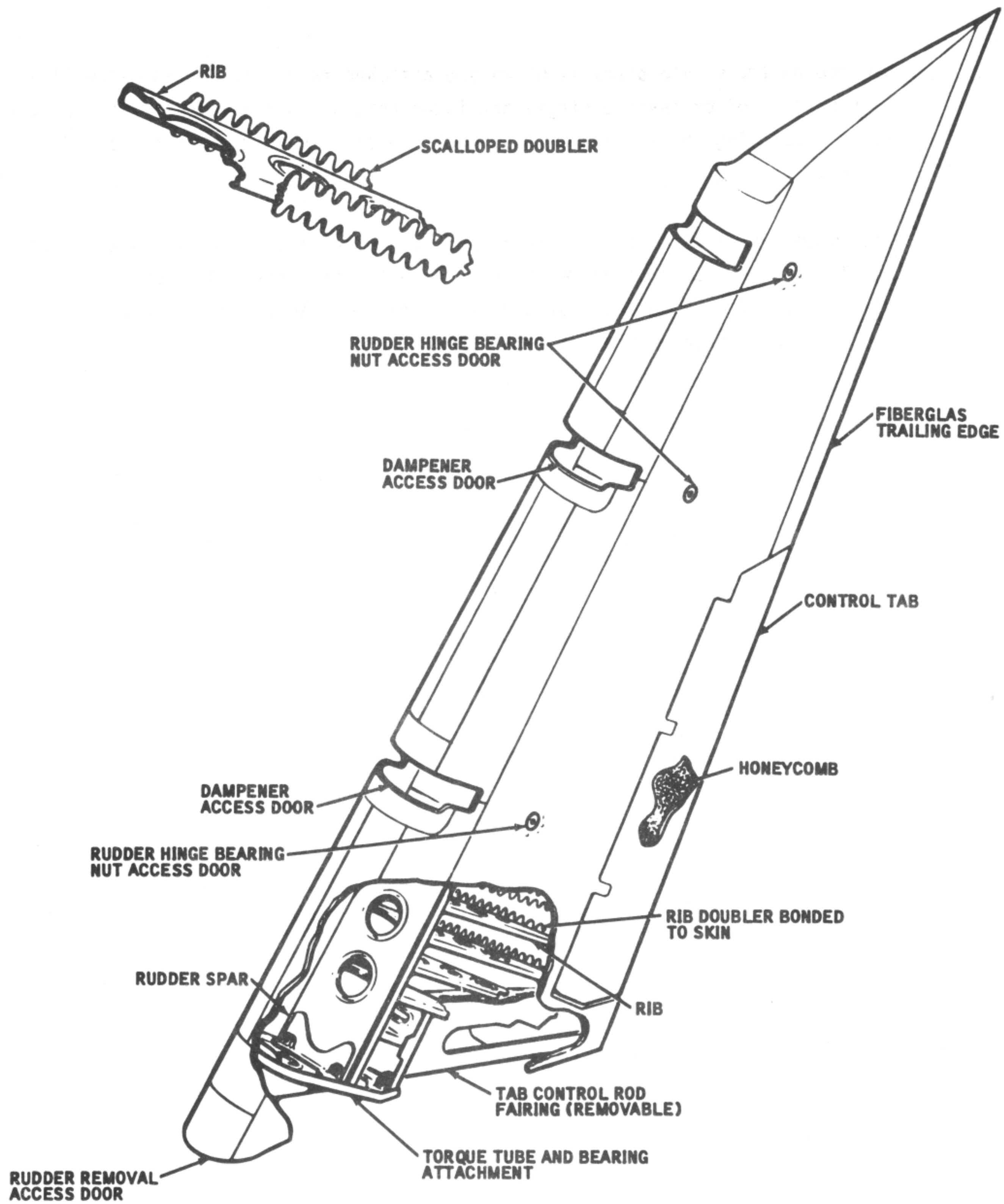


FIGURE 2-14. RUDDER STRUCTURE

mounted on truss-type brackets which are attached to the horizontal stabilizer rear spar. Roller bearing hinges are incorporated and bearings may be replaced without removing the elevator. A control tab and a geared tab are located in the trailing edge.

The rudder is an all-metal structure with one spar, ribs, and skin with bonded doublers. The rudder is sealed to prevent entry of water and excessive circulation of air, and is provided with drain openings. An all-metal control tab is incorporated in the trailing edge.

PYLON STRUCTURE

The pylons are located on each side of the aft fuselage (see Figure 2-15). These structures mount the engines and house fuel, electrical, hydraulic, heating, ice protection and control lines to and from the engines. The engine is housed by two cowl doors and a pylon apron.

The pylon consists of a main box, auxiliary structures and attach fittings. The main frame is the area of the pylon between the front and rear spar and is permanently attached to the fuselage. Auxiliary structures are the leading and trailing edges between the fuselage and pylon apron. Each engine is mounted through two sets of attach fittings. The pylon skin is titanium, except for a small section of the removable trailing edge which is aluminum and fiberglass.

The pylon is constructed to provide maximum fire protection. A firewall within the pylon follows an irregular path and extends from the leading edge to the fiberglass trailing edge. The firewall is constructed of titanium and all lines passing through it are provided with metal connectors. In addition to the firewall, a fire resistant barrier constructed of laminated phenolic fabric material and capable of withstanding maximum temperatures of 3000°F (1649°C) is bolted to the firewall in the center portion of the main frame area. (Further protection is provided by the skin of the fuselage, adjacent to the pylon, which is constructed of titanium material and forms a fireseal.)

The main frame of the pylon is of two-spar box construction and is permanently attached to the aft fuselage. The spars extend into the aft fuselage and attach to fuselage bulkheads with lockbolts and collars. Materials used in construction of the pylon are fire resistant except for nonstructural fairings.

The front spar is constructed of angles of 4140 steel, reinforced with strap corrosion-resistant 17-7PH steel, and capped with forged steel support fittings. The forward engine mount is attached to the support fittings. The rear spar is constructed of strap 4140 steel and has bearings in the outboard end for mounting the aft engine mount. The front and rear spar webs, stiffeners, and doublers are titanium and provide firewall protection within the pylon.

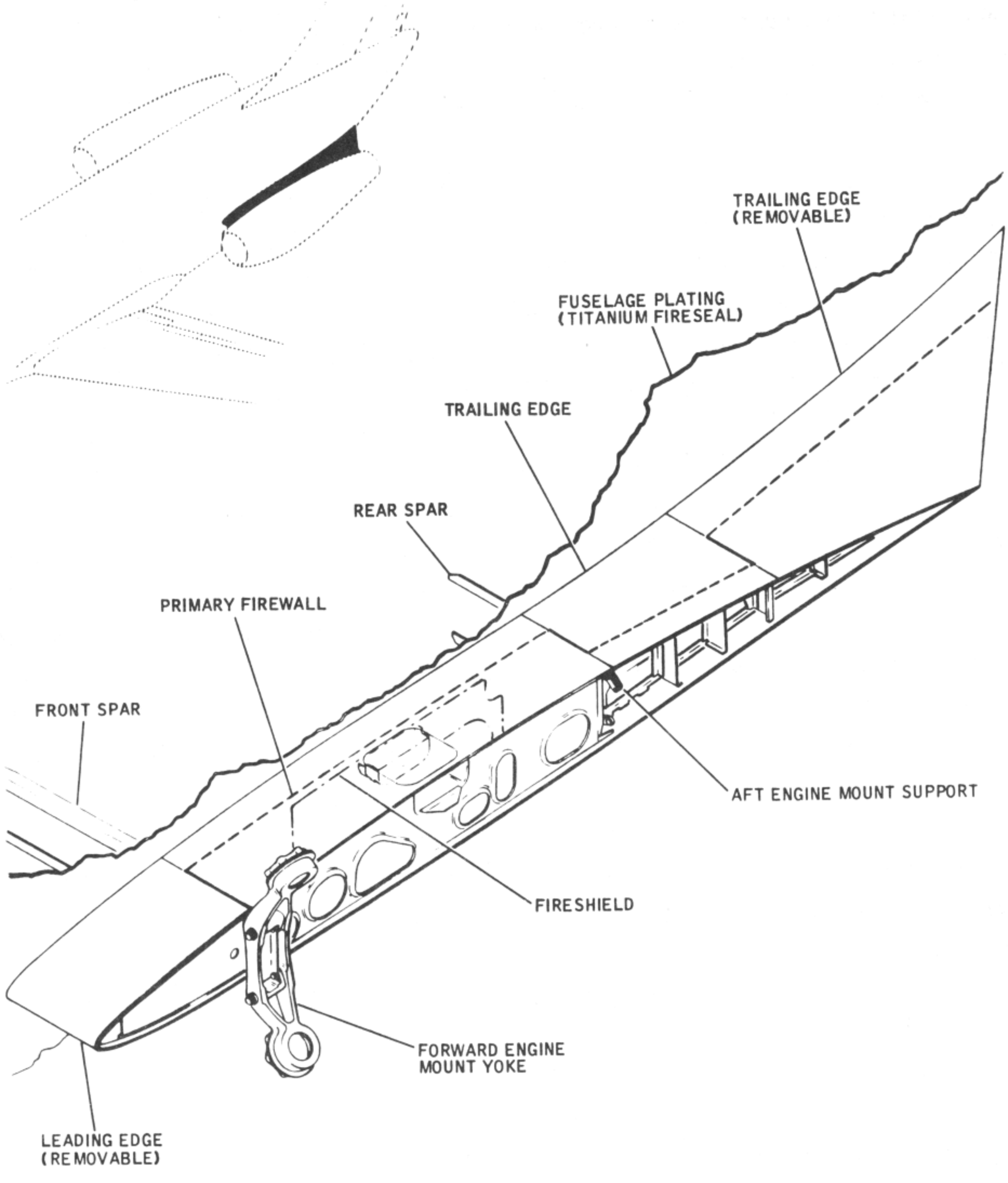


FIGURE 2-15. PYLON

There are two vertical ribs of titanium between the front and rear spars. These two ribs run parallel to the engine centerline and the inboard rib forms a firewall.

Access to disconnect engine fuel, pneumatic, and electrical system components is through the lower panel load-carrying access doors. High-shear, quick-action type fasteners are used to secure the doors. Access to the hydraulic systems component lines is through an access door, installed with flush screws, in the upper panel.

The auxiliary structure of the pylons, leading and trailing edges, completes the airfoil profile that supports the engine. The leading edge closing rib, skin, and stiffeners are titanium and the other ribs are aluminum alloy. The closing rib is the primary firewall in the leading edge. The leading edge is removable for inspection and maintenance and attaches to the front spar, fuselage-to-pylon, former and closing rib with flush screws.

The trailing edge is in two sections. The forward section skin and stiffeners are constructed of titanium. The removable aft section is constructed of aluminum alloy and fiberglass. A rib of titanium extends through the aft section for the primary firewall. The forward section has a quick-opening access door to provide access to the engine thrust reverser lines and aft engine isolator mount bolts.

POWER PLANT

The DC-9 is powered by two Pratt & Whitney JT8D axial flow turbofan engines. Each engine in the installed configuration is a demountable power plant.

The demountable power plant is connected to a short, horizontal pylon by a three-point mounting system which utilizes engine-mounted cone bolts, and pylon-mounted vibration isolators. This mounting system minimizes the transfer of engine vibration to the main airplane structure.

The power plant cowling sections join together around the engine and accessories providing smooth fairing with the pylon. This provides complete power plant protection and unrestricted airflow during flight.

Each power plant, after installation on the airplane, is protected by a fire detection system and a fire extinguishing system.

FUEL SYSTEM

The airplane fuel system consists of fuel storage, fuel fill and defueling, fuel tank vent, fuel distribution, and fuel quantity indication. Fuel is stored in three integral tanks: left main, right main, and center. In addition, some configurations have fuel cells in the fuselage belly. A series of flapper valves incorporated into two flow baffles near the inboard end of each tank breates a reservoir. The reservoir provides a head of fuel around the boost pumps during all normal maneuvers and airplane attitude changes. Each main tank is provided with an overwing fill adapter and a sump drain valve. The center tank is provided with two sump drain valves.

Fuel feed is accomplished by means of two AC boost pumps in each tank. Pumps for the center tank are located in a container in the right wing reservoir. The main tank pumps are installed in individual volutes. The two center tank pumps are connected in series in a single volute. A DC start pump is located in the right tank reservoir and connected to the feed line. The pump supplies fuel pressure for engine and APU starting when AC power is not available. The right and left main tank-to-engine fuel feed lines are interconnected by a crossfeed line and a manually operated crossfeed valve. One center tank boost pump feed line connects to the crossover section of the crossfeed line downstream of the crossfeed valve to supply center tank fuel to the left engine. Check valves in the center tank pressure lines prevent reverse flow of fuel when the main tank pumps are operating. These two check valves are spring loaded to approximately 1 1/2 psi to prevent the engine-driven pump from pumping air from an empty center tank if the main boost pumps are off or have failed.

A manually actuated engine fuel fire shutoff valve is located in the engine supply line on the aft face of the wing rear spar. A solenoid shutoff valve is located in the APU supply line. The valve is mounted next to and inboard of the right engine fuel fire shutoff valve. All fuel lines passing through a pressurized area are enclosed in shrouds. The shroud lines are vented through ports on each side of the fuselage aft of the APU inlet. A shroud drain valve is located on the right lower side of the fuselage aft of the wheel well (see Figures 2-16 and 2-17).

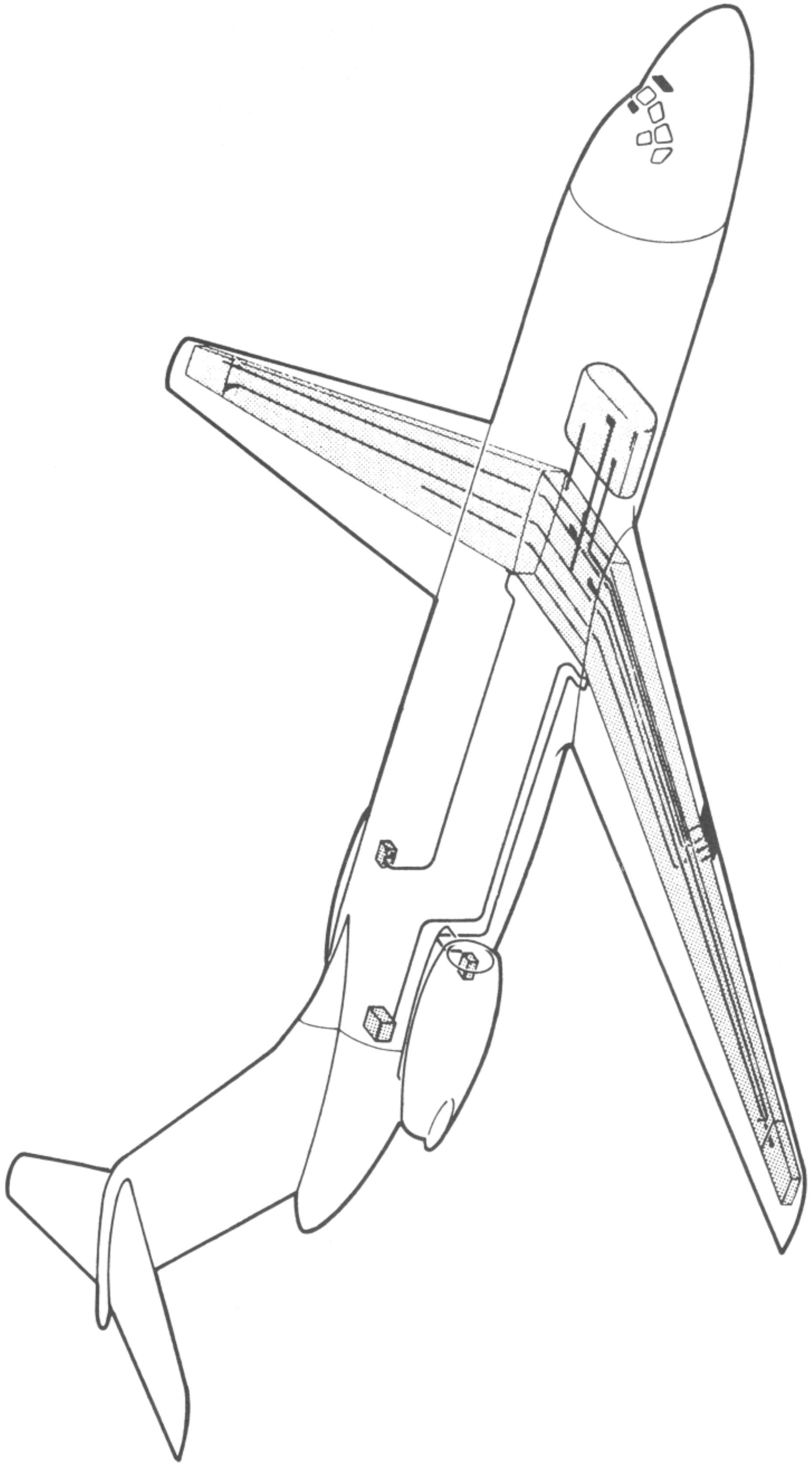


FIGURE 2-16. FUEL SYSTEM ARRANGEMENT — 4 TANK

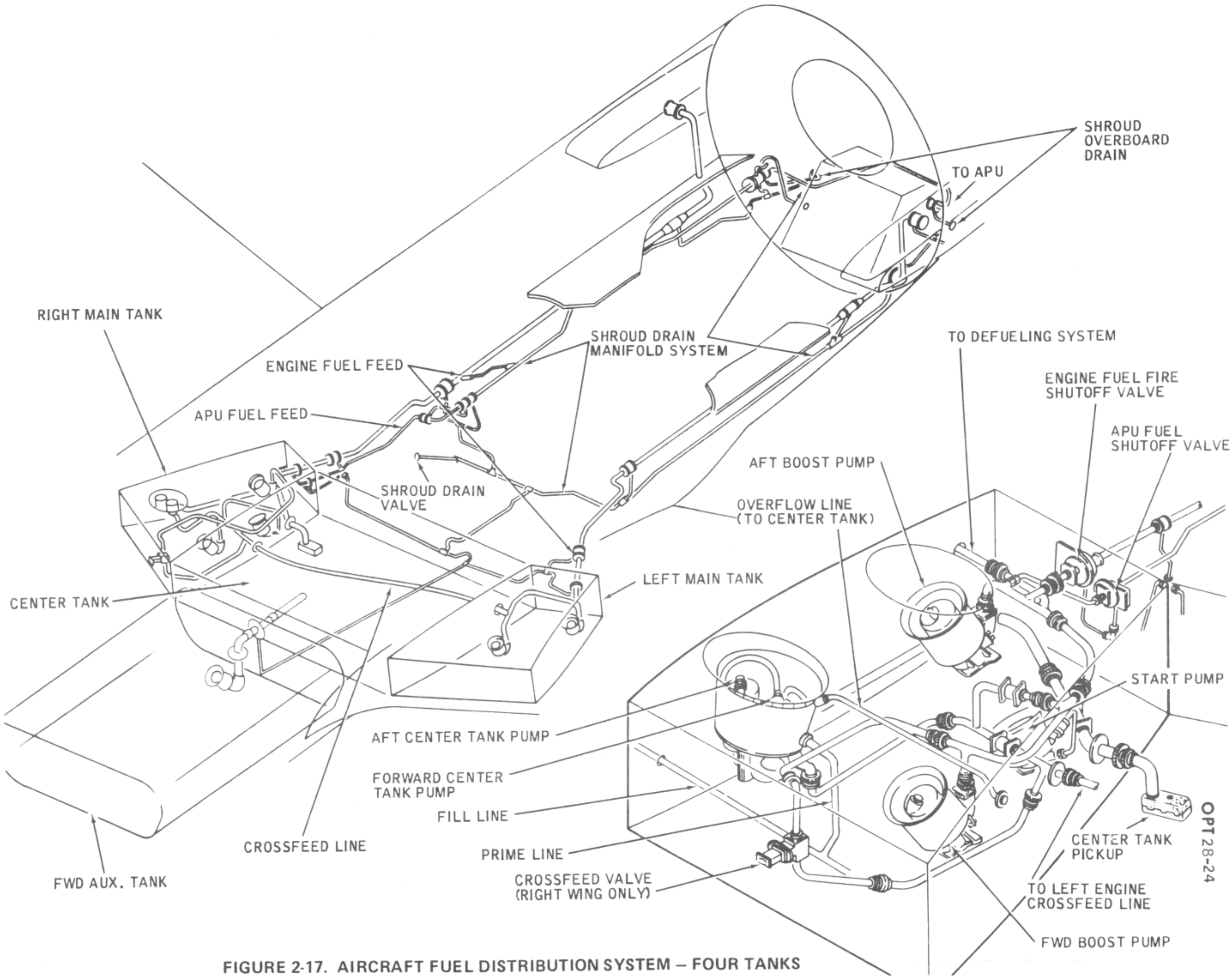


FIGURE 2-17. AIRCRAFT FUEL DISTRIBUTION SYSTEM – FOUR TANKS

FLIGHT CONTROLS

The flight controls consist of the control surfaces, trim control surfaces, indicating and warning systems, and the related mechanical, hydraulic, and electrical systems that control the airplane during flight and on the ground.

The control surfaces and their locations are listed as follows:

Control Surface	Location
Ailerons	Outboard trailing edge of each wing
Elevators	Trailing edge of horizontal stabilizer
Rudder	Trailing edge of vertical stabilizer
Flaps	Trailing edge of each wing, inboard of ailerons
Flight spoilers	Trailing edge of each wing, on upper surface inboard of ailerons
Ground spoilers	Trailing edge of each wing, on upper surface inboard of flight spoilers

The ailerons provide lateral control of the airplane and are operated by aerodynamic boost control surfaces. A control tab, on the trailing edge of each aileron, is deflected and aerodynamic forces on the tabs move the aileron. The control tabs are controlled mechanically by the dual aileron control wheels in the flight compartment (see Figure 2-18).

The elevators provide longitudinal control of the airplane and are operated normally by aerodynamic boost control surfaces which consist of a control tab on the trailing edge of each elevator. The control tab is controlled mechanically by dual control columns in the flight compartment (see Figure 2-19). A geared tab on each elevator trailing edge is linked mechanically to the horizontal stabilizer so that as the elevator moves, the geared tab is moved in the same direction as the control tab to provide additional aerodynamic control force. If both control tabs are deflected 10 degrees or more for downward movement of the elevator, an elevator power (hydraulic) control mechanism is operated to assist in driving the elevator down.

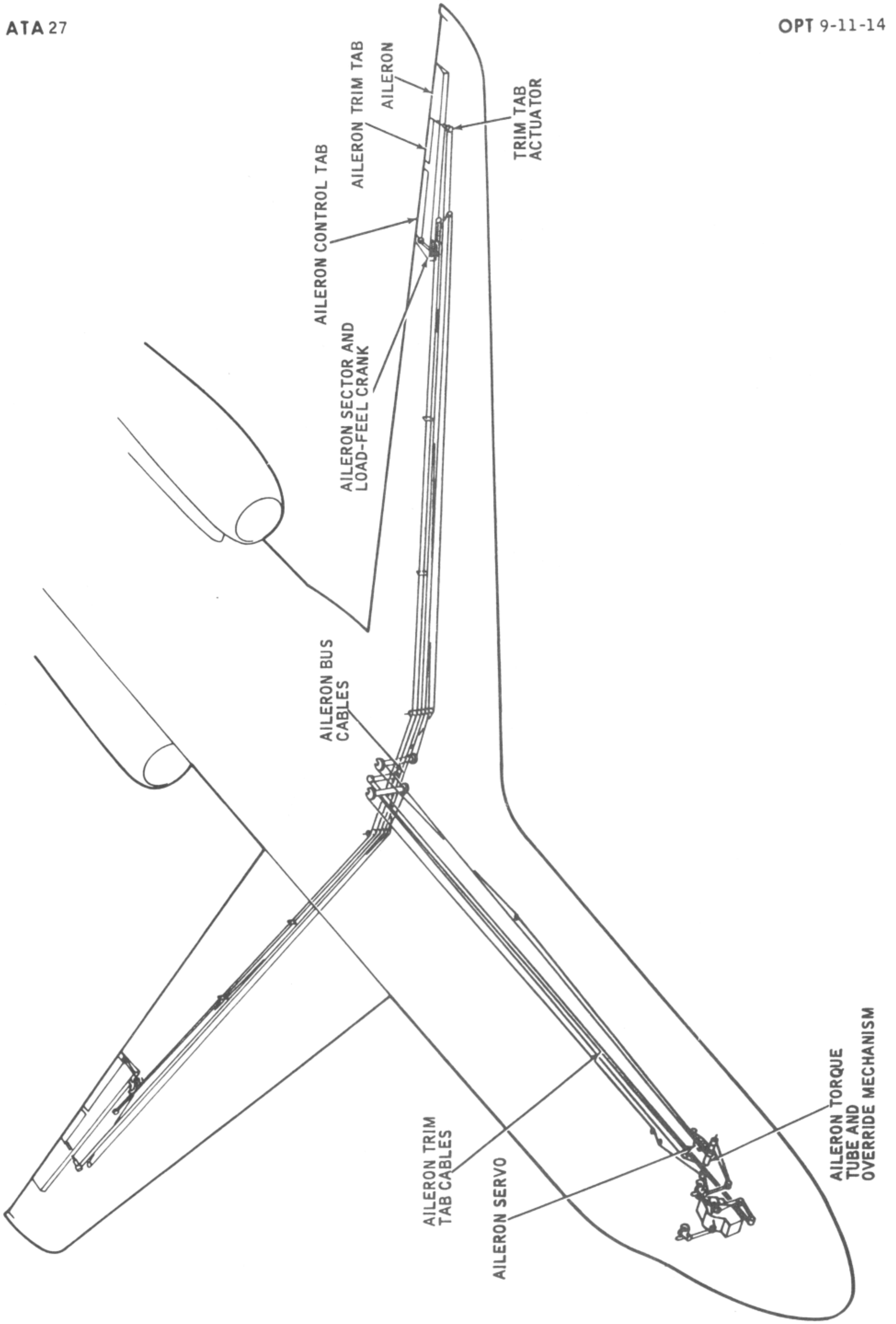
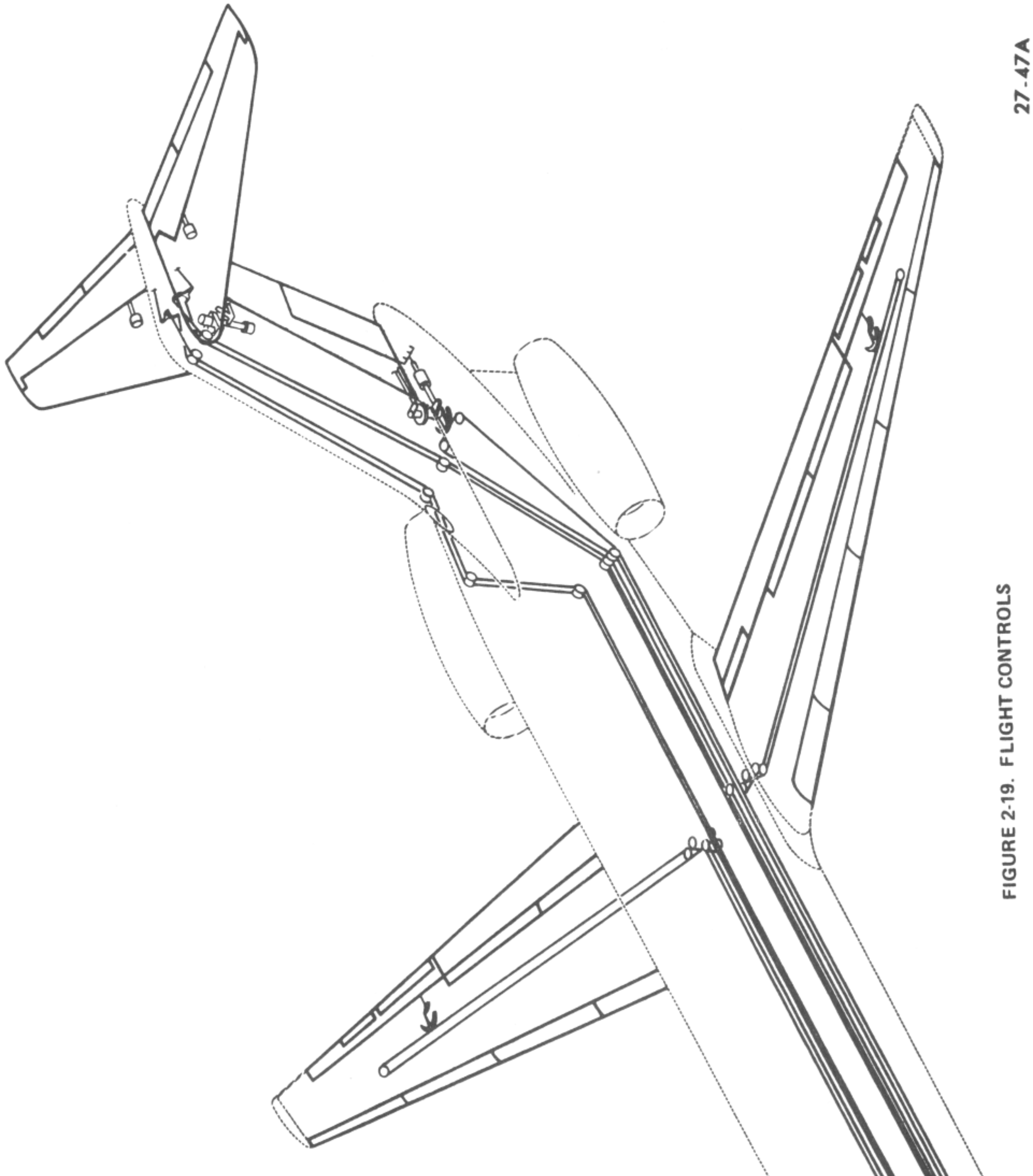


FIGURE 2-18. AIRERON AND TRIM CONTROL SYSTEM



27-47A

FIGURE 2-19. FLIGHT CONTROLS

The rudder provides directional control of the airplane about the vertical axis, and is actuated hydraulically. The hydraulic components are controlled mechanically by dual rudder pedals in the flight compartments(see Figure 2-20). The control tab on the rudder trailing edge is locked hydraulically in faired position during hydraulic operation. If hydraulic pressure drops below 950 (±150) psi, the control tab is unlocked and becomes an aerodynamic boost control surface operated mechanically by the rudder pedals. Manual operation may also be selected by closing a hydraulic power shutoff valve, controlled by a lever on the control pedestal. Maximum rudder travelis 37 degrees hingewise left or right of faired position, but as airspeed increases, rudder travel is progressively decreased by a limiting device until at 330 knots, travel is restricted to 3-1/2 degrees in each direction.

The flaps increase the lift of the wing when partially extended during takeoff or landing approach, and increase drag to reduce speed when fully extended during landing. The flaps are actuated hydraulically and the hydraulic components are controlled mechanically by the flap control lever on the control pedestal.

The flight spoilers supplement the ailerons in lateral control and also provide a speedbrake during flight. The flight spoilers are actuated hydraulically and when used as a lateral control aid, are controlled by the aileron control wheels. When used as a speedbrake, the hydraulic components are controlled by the speedbrake control lever on the control pedestal. Flight spoilers are also used with the ground spoilers during the landing roll.

The ground spoilers supplement the flaps by decreasing lift to reduce speed during the landing roll. The ground spoilers are actuated hydraulically, controlled electrically, and may be set to operate automatically or may be operated manually by the speedbrake control lever. When the control lever is set in the armed position, the ground spoilers and the flight spoilers in the speedbrake mode extend automatically when the main gear wheels spin at 700 rpm after contact with the ground.

The aileron trim tabs and the horizontal stabilizer are adjustable trim control surfaces that provide a means of adjusting the aerodynamic characteristics of the main control surfaces.

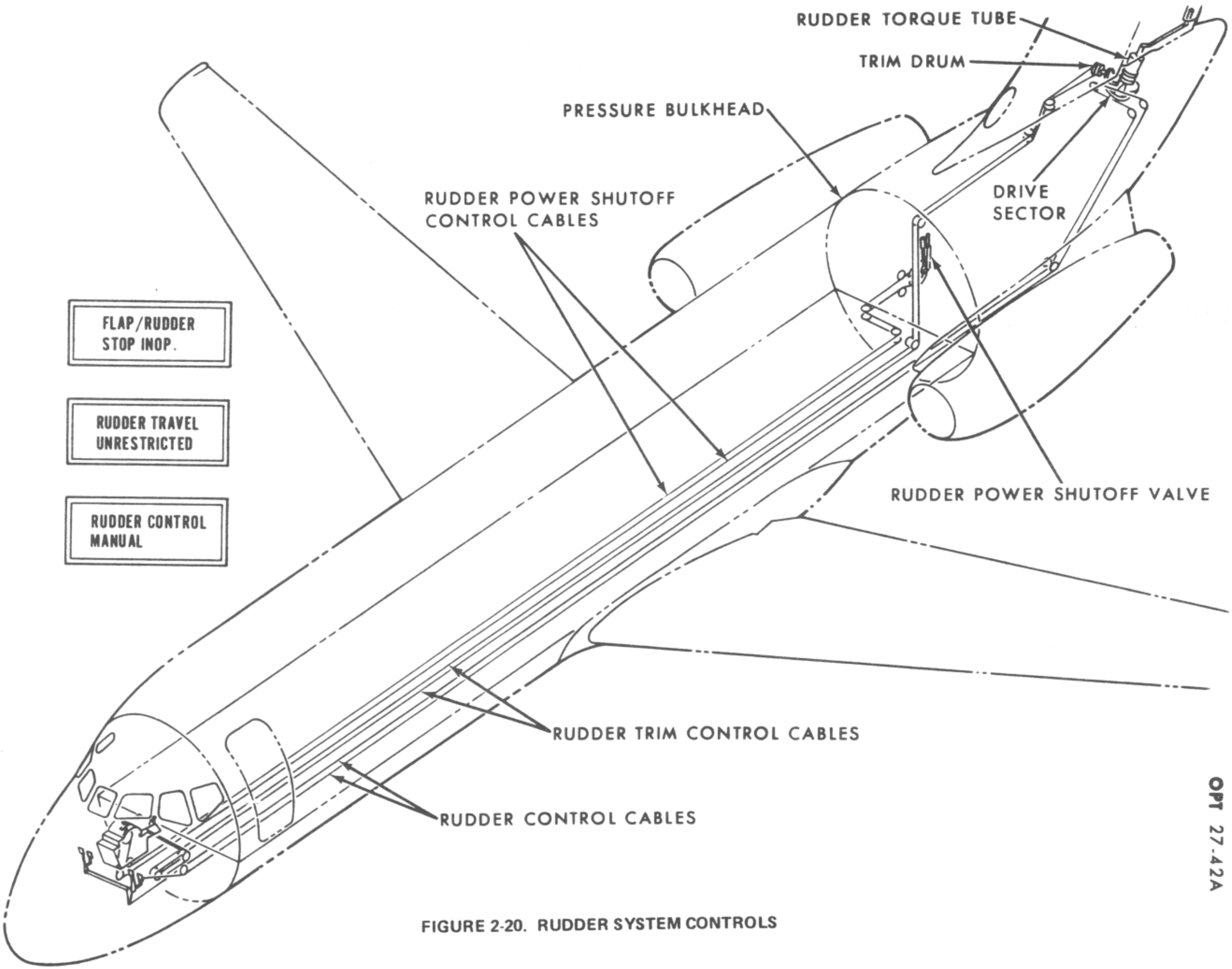


FIGURE 2-20. RUDDER SYSTEM CONTROLS

The aileron trim tabs are controlled mechanically by cable systems from the aileron trim control knob on the control pedestal. As the tab on one aileron is moved up, the tab on the opposite aileron is moved down to provide lateral trim.

Rudder trim is obtained by rotating the rudder trim control knob on the control pedestal. This moves the rudder trim and load-feel mechanism the rudder pedals to a new neutral position. During power operation the hydraulic components move the rudder left or right in the amount of trim selected. During manual operation, the control tab on the rudder is moved so that aerodynamic forces on the tab move the rudder to the selected trim position.

The horizontal stabilizer provides longitudinal trim and is controlled by the longitudinal trim control system. The system is actuated electrically and may be controlled by dual control handles on the control pedestal or by switches in the aileron control wheels. An alternate system of moving the horizontal stabilizer is actuated electrically and controlled by two levers on the control pedestal.

Indicating and warning systems are used to show the position of the trim control surfaces and provide warning when certain controls are being used or should not be used.

Mechanical indicators on the control pedestal show the amount of trim for the ailerons, rudder, and the horizontal stabilizer. A remote electrical indicator, on the center instrument panel, indicates the position of the flaps.

Indicator lights, on the annunciator panel in the flight compartment, show when the rudder is in the manual mode of operation, when rudder travel is unrestricted, and when not to use the ground spoilers in the automatic mode.

A takeoff warning system sounds an intermittent audible warning if the throttles are advanced for takeoff when the horizontal stabilizer, flaps or spoilers are not in proper position for takeoff.

A landing gear warning system sounds a continuous audible warning if the flaps are extended past the approach position (approximately 28 degrees of flap extension) with the landing gear not down and locked.

A stabilizer motion warning is sounded when the horizontal stabilizer is being moved by either of the three methods of control or by the autopilot trim. The system provides a dual tone signal once for approximately each 1/2 degree of stabilizer movement.

HYDRAULIC POWER

Hydraulic power is provided by two hydraulically separate, closed-circuit hydraulic systems identified as the left system and the right system. Fire-resistant Skydrol hydraulic fluid is used in both systems. The two systems are mechanically interconnected by the alternate gear pump which consists of an electrically controlled, reversible, hydraulic motor-pump. The two systems are basically similar except for component location and subsystems served. The right system provides hydraulic power to the rudder power subsystem and the landing gear actuating subsystem; the left system provides hydraulic power to the ground spoiler subsystem and the elevator power subsystem. All other subsystems are served by both hydraulic systems through separate valves and actuators.

Each system is equipped with a hydraulic fluid supply reservoir and an engine-driven hydraulic pump which normally provides sytem power. An electrically driven (auxiliary) hydraulic pump is provided in each system for backup of the engine-driven pumps. The alternate gear pump provides a second level backup for the left engine-driven and electrically-driven hydraulic pumps. It also provides backup in case of total depletion of the normal fluid supply in the right system, by providing pressure to actuate the landing gear by mechanical connection with the left hydraulic system pressure. In this case, an emergency supply of fluid entrapped in a sump in the right system is utilized.

The engine-driven hydraulic pumps are mounted on the accessory gear cases of their respective engines. Each engine-driven pump is equipped with a solenoid-operated depressurization valve which, when energized, allows a spring-loaded blocking valve to close and shuts off hydraulic pressure to the system, feathers the pump, and ports pump leakage and lubrication fluid through the case drain line and filter, and through the system return line filters to the reservoir. The engine-driven pumps are also equipped with solenoid-operated unloading valves to reduce pump output pressure to approximately 1500 psi when full system pressure (3000 psi) is not required during cruise flight.

Pressurized fluid from the engine-driven pumps is directed through pressure line filters of the general system. Pressure relief valves are provided in the same area to relieve engine-driven pump output pressure at 3400 (±50) psi.

A mechanically operated fire shutoff valve is provided in each hydraulic system. The fire shutoff valve is actuated only in the event of fire in the engine compartment and stops all flow of hydraulic fluid to the engine-driven hydraulic pump.

The electrically driven (auxiliary) hydraulic pumps are located in the forward area of the main gear wheel wells, and are controlled (on and off) by left and right auxiliary hydraulic pump switches in the flight compartment. They are capable of supplying a continuous flow of hydraulic fluid at 8 gpm and 2200 (±50) psi or 6 gpm and 2750 (±50) psi. The motors are operated on 115-volt, 3-phase, 400-cycle power supplied by the opposite AC generator bus.

An accumulator is installed in each of the two hydraulic power systems to provide a reserve supply of fluid under pneumatic pressure and to act as shock absorbers for system pressure surges.

Overpressure protection is provided by hydraulic power system pressure relief valves, one for each system, which start to relieve at 3400 (±50) psi and return excess pressure to the system return lines and reservoirs.

The alternate gear pump which mechanically connects the two hydraulic systems consists of two axial piston hydraulic units set back-to-back and, connected together by a shaft, the two units being completely separate hydraulically. The units are constructed so that either side can operate as a motor while the other operates as a pump. The units are controlled by solenoid-operated, two-position (open - closed) shutoff valves, one valve located in each motor-pump pressure line. The valves are controlled simultaneously by the alternate gear pump switch in the flight compartment which energizes the valve solenoids to open the valves. With one system pressurized and the other depressurized, the unit of the motorpump located in the pressurized system automatically becomes a motor, while the unit in the depressurized system becomes a pump when the solenoid valves are opened. The pump unit thereby provides pressure to the depressurized system. Flow control valves, rated at 2 gpm, are provided in the alternate gear pump pressure lines to restrict the outlet flow of the pumping unit of the alternate gear pump to prevent cavitation of the suction side of the pump. The right system has a fluid supply sump located in series with the landing gear control valve return line, which traps sufficient fluid to provide an alternate supply for motorpump actuation of the landing gear and main gear inboard doors in case of total depletion of the normal supply in the right system.

PNEUMATIC SYSTEM

The penumatic system consists of two identical subsystems, each complete and operable independently of the other or interconnected to provide a common pressure source for the using systems (see Figure 2-21).

Engine bleed air from the 8th-stage (low pressure) compressors and 13th-stage (high pressure) compressors is bled off through bleed ports into separate bleed air manifolds and into the pneumatic system ducting.

Hot air from the pneumatic ducting is supplied to the air conditioning system, the ice protection system, and the engine starting system.

The left and right systems each contain a crossfeed valve, which isolated the interconnecting ducting from the supply ducts. The right engine normally supplied engine bleed air to the right system upstream from the right crossfeed valve; the left engine supplies the left system upstream from the left crossfeed valve. The auxiliary power unit (APU) or a ground pneumatic source can also supply pressure to both the left and right systems.

The pneumatic system is divided into the distribution and the indicating systems. The distribution system distributes pneumatic pressure to the using systems. All components of this system are located in the engine area, pylons, and aft accessory compartment. The indicating system consists of pressure indicating and overheat warning components. Visual indication of system pressure is provided by the pneumatic pressure indicator, which is connected electrically to the pressure transmitter. The indicator is located on the forward overhead switch panel in the flight compartment, and the transmitter sense line connects to the crossfeed ducting, between the right and left crossfeed valves. Correct system pressure will not be displayed on the pneumatic pressure indicator with the engines operating, unless a crossfeed valve is open.

If temperature exceeds 180 ($\pm$10)^{0}F in the aft accessory compartment, the tail compartment temperature high caution annunciator in the overhead annunciator panel, and the master caution lights in the glareshield will come on.

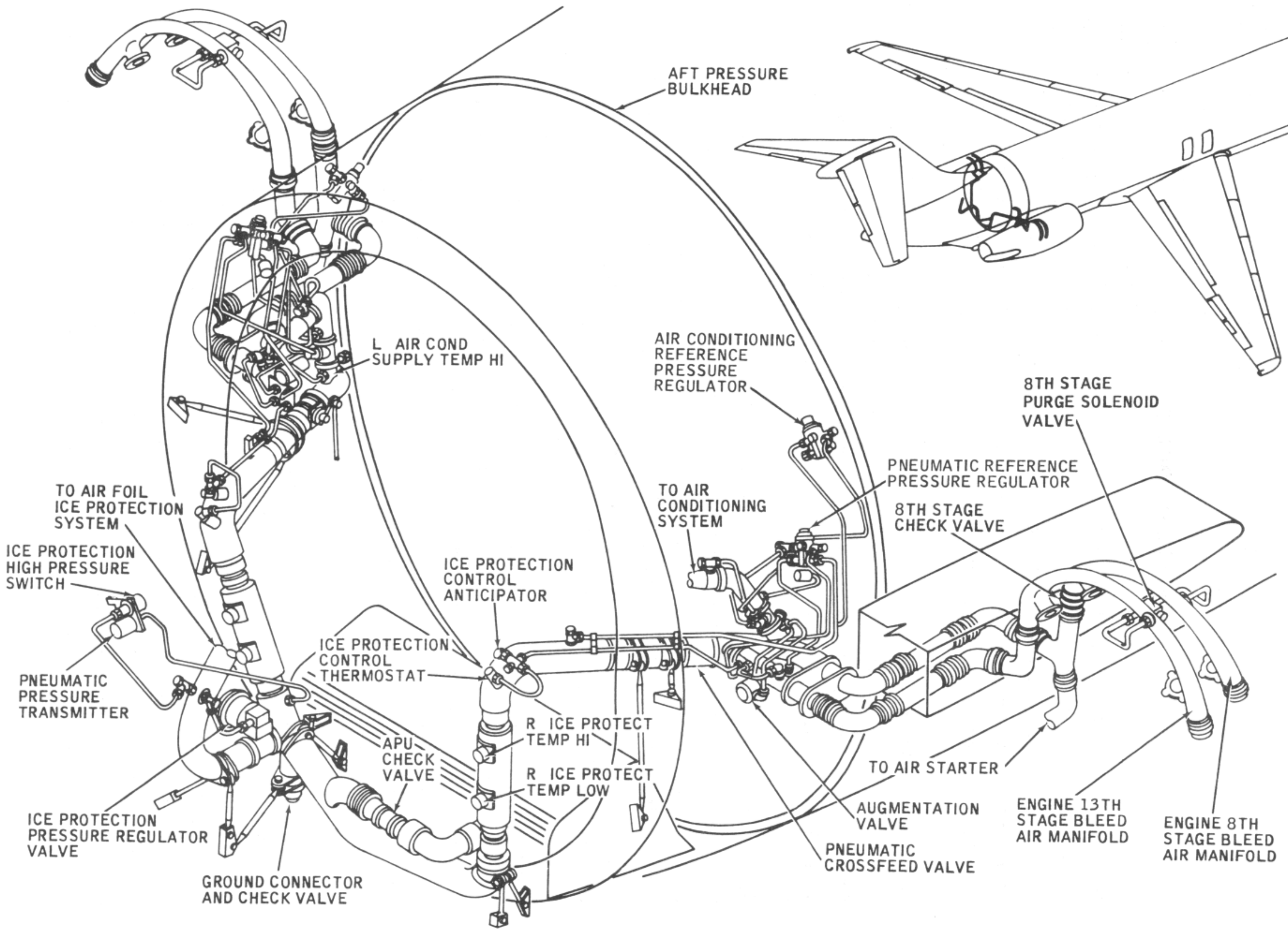


FIGURE 2-21. PNEUMATIC SYSTEM GENERAL LAYOUT

AUXILIARY POWER UNIT

The auxiliary power unit (APU) supplies pneumatic and electrical power for ground starting of main engines, fuselage air conditioning, and ground operation of the airplane electrical system in the absence of ground power or an operating engine. The APU can also be operated in flight to provide an alternate source of electrical power.

The APU is mounted in a fireproof compartment in the lower aft portion of the fuselage, aft of the pressure bulkhead (see Figure 2-22). Two large doors located at the bottom of the fuselage permit access to the APU. The demountable APU is attached to the airplane by four adjustable mounts.

Air for the gas turbine is inducted through a set of electrically controlled air inlet doors located on the lower surface of the APU demountable support box. The air induction system consists of a ram air door, two non-ram doors and an electrically driven actuator controlled by a switch located in the flight compartment. Protection against ingestion of foreign objects is provided by a screen attached to the turbine inlet.

The APU is a single-shaft, two-bearing, gas turbine engine, containing a two-stage centrifugal compressor, a single combustion liner, a single-stage radial inflow turbine, and accessory drive. The APU is provided with the following systems: engine fuel and control, ignition/starting, air, engine controls, indicating, exhaust, and oil. In addition, fire detection and fire extinguishing systems are provided for the APU.

The engine control system provides means for controlling APU operations on the ground or in flight. Two separate control panels are provided: the master control panel and an external control panel. The master control panel is located on the overhead switch panel in the flight compartment and is used for starting and stopping the unit, and for positioning the air inlet door. Fire detection system operation and test, and fire extinguishing system operation are controlled from this panel. In addition, controls for application and removal of pneumatic and electrical power are provided on the

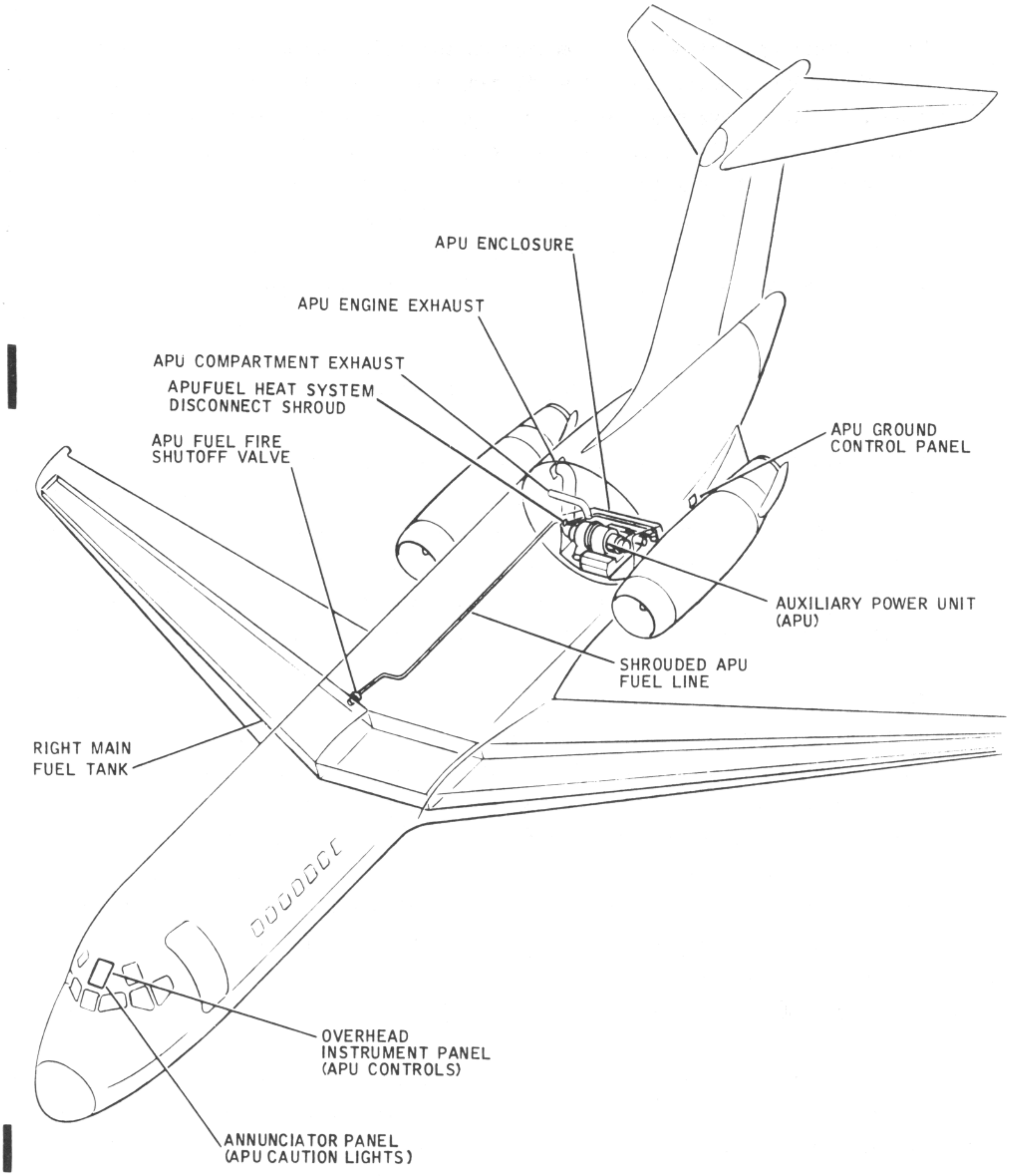


FIGURE 2-22. AUXILIARY POWER UNIT – LOCATION

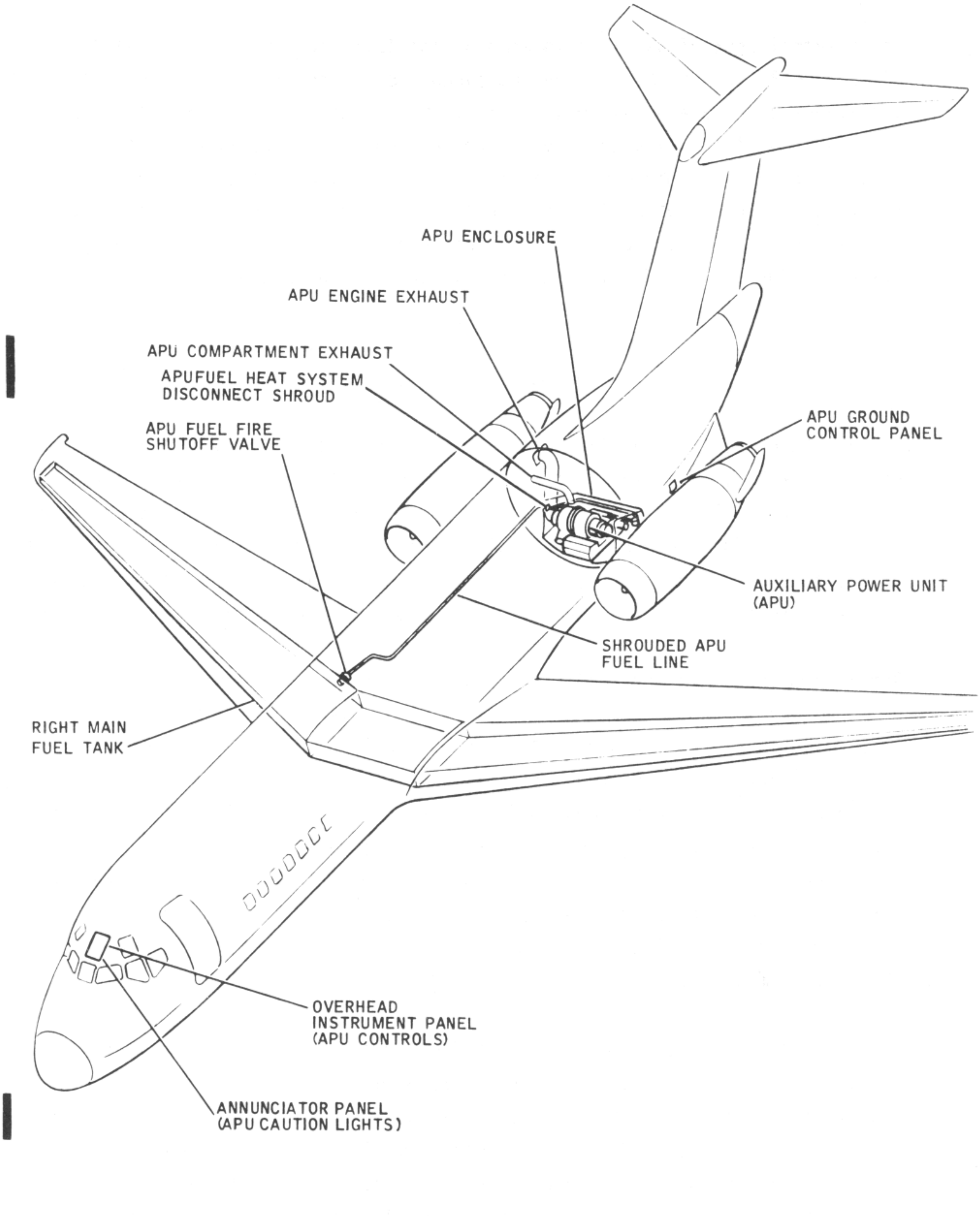
The external control panel is located on the left side of the airplane just aft of the APU compartment. The APU cannot be started from controls on this panel, but there are provisions for stopping the unit. The panel is equipped with fire warning indicators and a fire warning horn. The fire extinguishing system for the APU compartment can also be controlled from this panel. The APU will shut down automatically when the fire warning system is activated. A centrifugal switch, mounted on the engine accessory drive, will automatically shut down the unit under conditions of overspeed.

AIR CONDITIONING SYSTEM

The airplane has two identical air conditioning systems designed for independent or parallel operation to supply conditioned air, cold air, and pressurized air at a controlled volume and pressure. Normally, the right system operates from right engine bleed air and supplies the passenger compartment requirements. The left system operates from left engine bleed air and supplies flight compartment requirements. Either system can supply the needs of both compartments.

The flight and passenger compartments are the only areas directly supplied with pressurized, conditioned, and cold air. Pressurized areas and compartment locations are shown in Figure 2-23. The forward lower cargo compartment is heated by electrical/electronic compartment exhaust air and the aft cargo compartment by passenger compartment exhaust air. Heating is accomplished by flowing air through the gap between the cargo compartment floor and airplane skin insulation. There is no continuous airflow through the cargo compartments. Cargo compartment pressure equalization valves are installed in the ceiling of each cargo compartment to maintain pressure within the compartments equal to passenger compartment pressure.

Bleed air is furnished from the engine low pressure (8th-stage) or high pressure (13th-stage) bleed air manifolds, depending on engine power settings and demands on the pneumatic supply. When necessary, manually controlled crossfeed valves, located in the pneumatic supply ducts, allow bleed air from either engine to supply the needs of both air conditioning systems. Either or both air conditioning systems can be operated by supplying hot pneumatic



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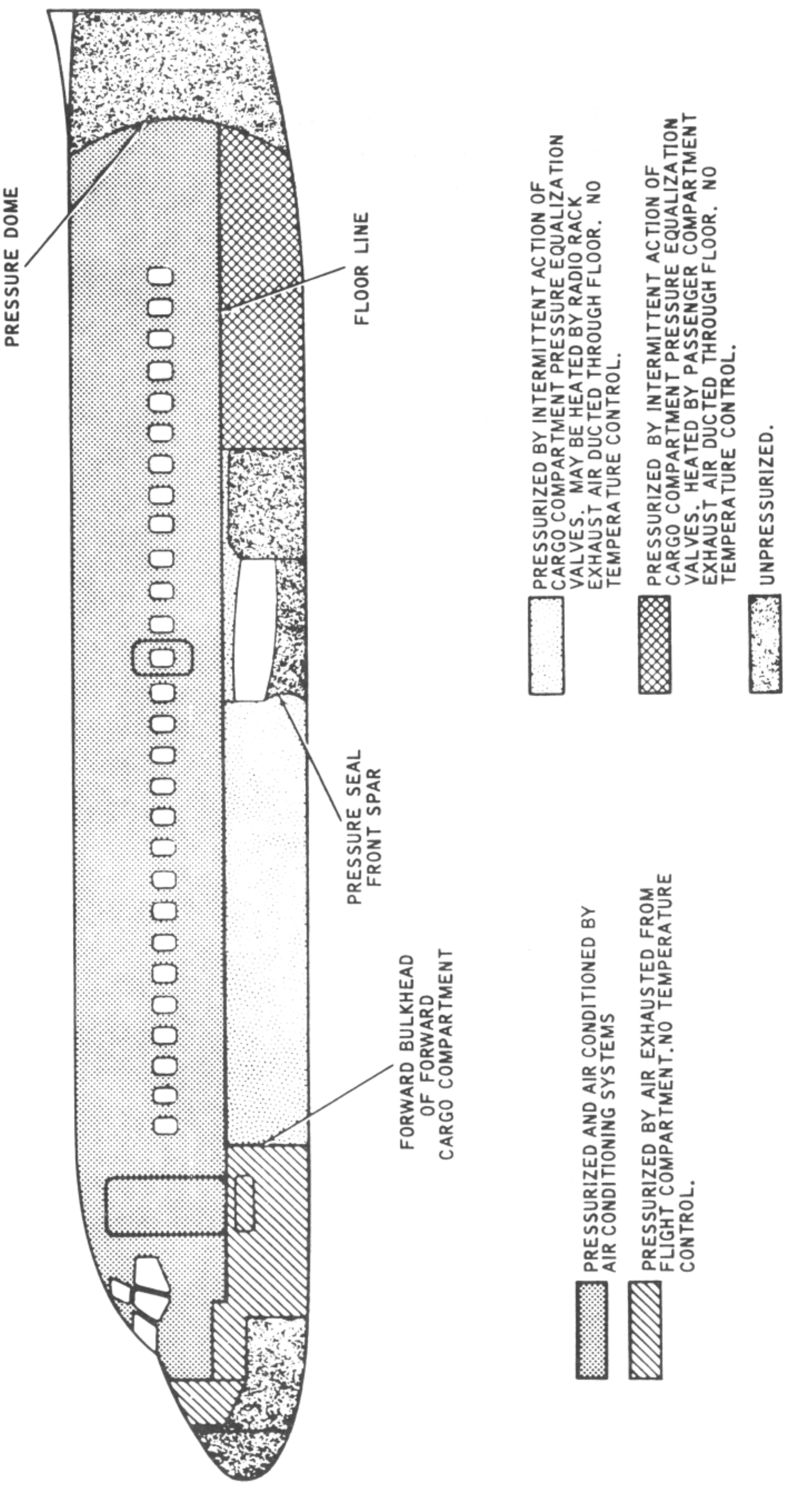

FIGURE 2-23. AIR CONDITIONED AND PRESSURIZED AREAS

pressure and electrical power from the APU (auxiliary power unit) in the airplane, or from a ground pneumatic supply connected to the pneumatic manifold and electrical power applied to the external pwoer receptable.

Although each air conditioning system supplies pressurized air, conditioned air, and cold air when the air conditioning system is turned on, each function of the system is controlled and monitored by separate subsystems. To segregate these functions, air conditioning is divided into distribution, pressurization control, cooling, and temperature control. Figures 2-24 and 2-25 show the locations of major components for the entire air conditioning system.

Conditioned air from both air conditioning systems is routed into a single cabin supply air mixer. Flight compartment air is ducted directly to that compartment for distribution through the various outlets. Passenger compartment air is ducted by a main feed to droppers which supply fore and aft dispersing ducts at the overhead stowage racks. The air enters the cabin through grills in the side of the dispersing ducts. Cold air is ducted to individual outlets above each passenger seat, and to outlets in the flight compartment, galley area, and lavatories.

Pressurization control permits selection of a desired cabin rate of altitude change to a predetermined cabin altitude. Both structural and comfort considerations apply limitations on the selection. The structural limitations permits establishment of a sea level cabin altitude up to flight altitude of 18,000 feet and a cabin altitude of 8000 feet up to a flight altitude of 35,000 feet. Rate selections of between approximately 50 and 750 feet per minute can be made.

The cabin air outflow valve provides cabin pressurization by controlling the area through which cabin air is exhausted to ambient. Normal pressurization during flight is automatic, but a manual control lever, which also indicates the position of the cabin air outflow valve, is provided to manually control cabin pressurization when necessary.

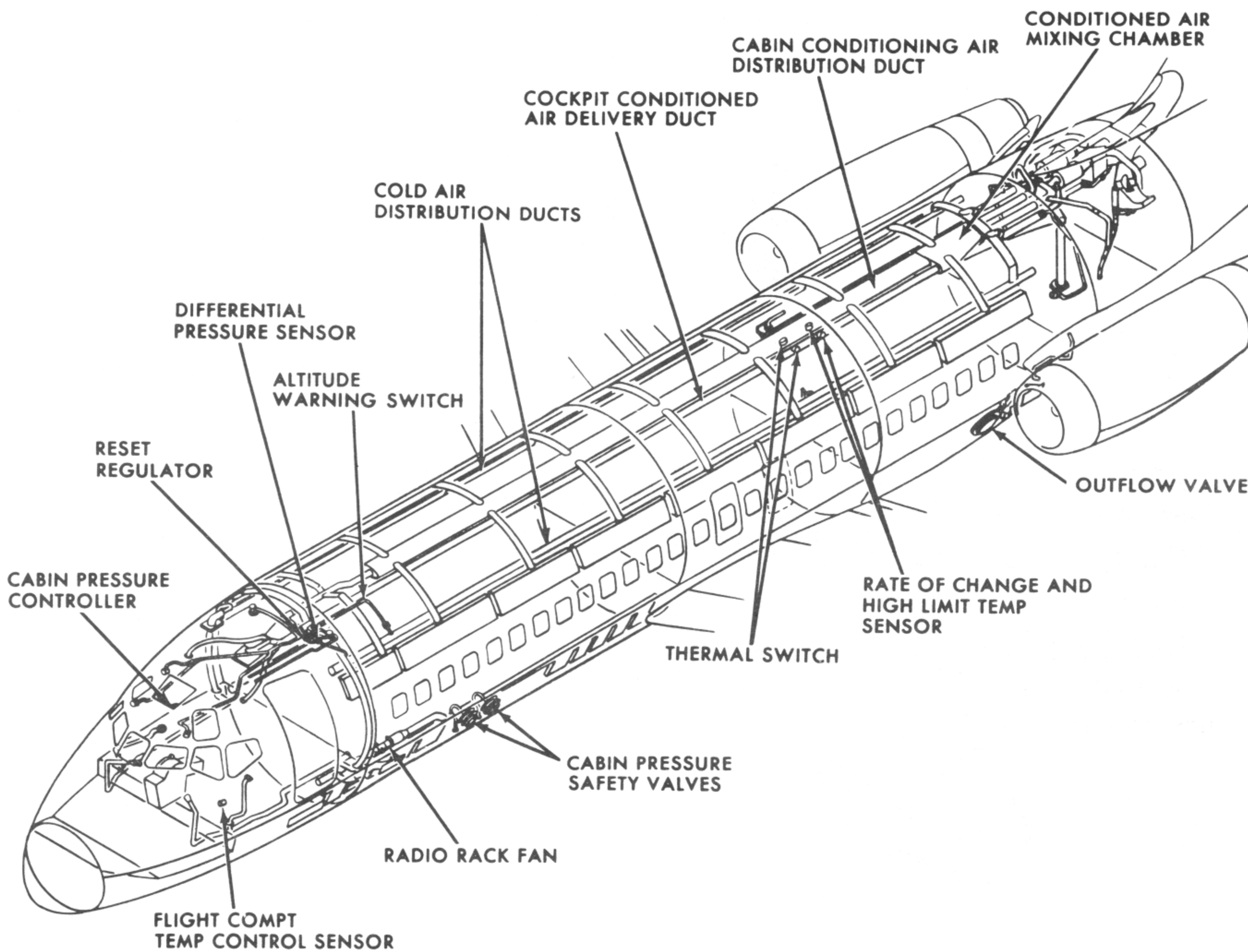

FIGURE 2-24. CONDITIONED AIR DISTRIBUTION

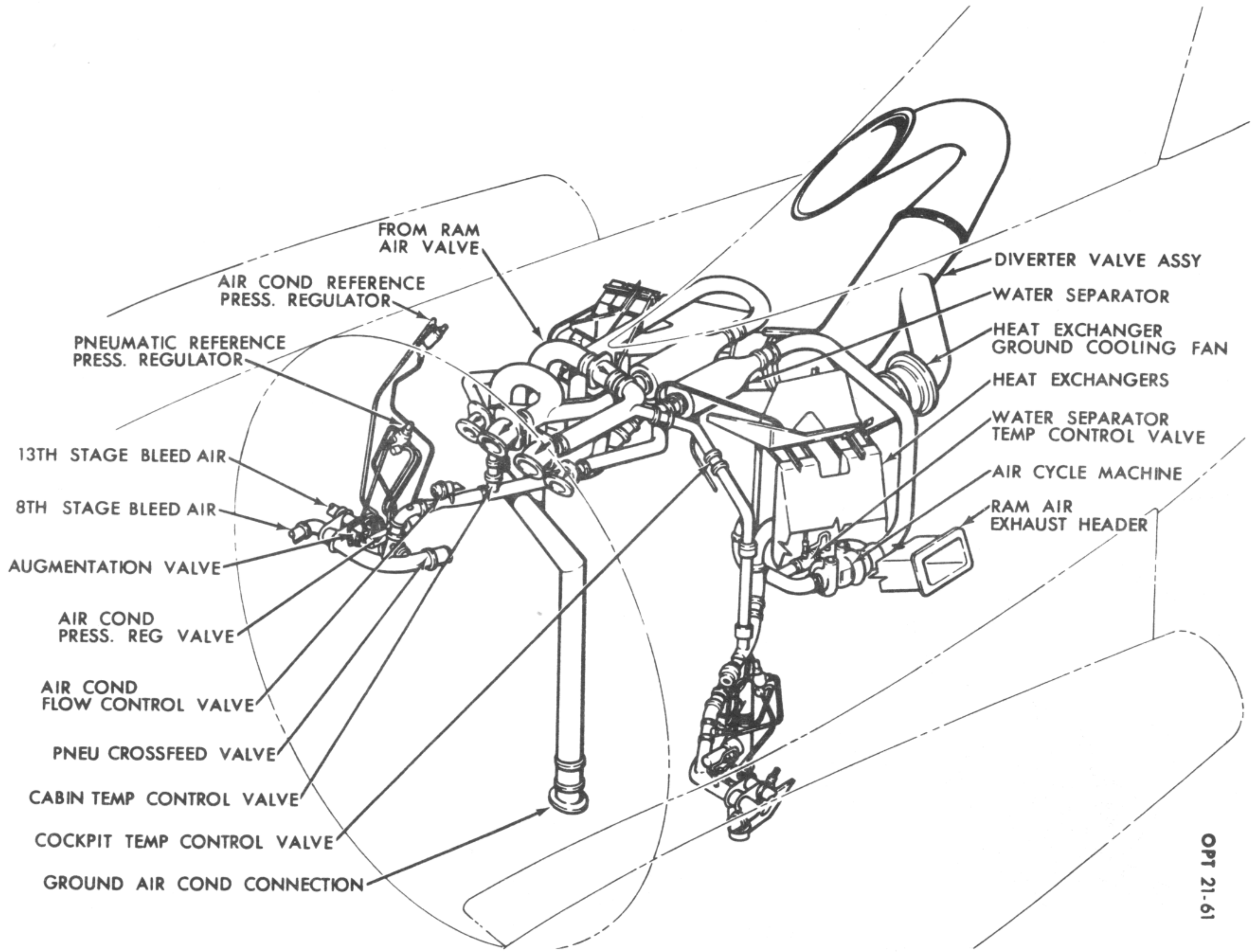

FIGURE 2-25. AIR CONDITIONING SUPPLY SYSTEM – GENERAL LAYOUT

Two cabin pressure safety valves prevent cabin pressure from exceeding a cabin-to-atmosphere differential pressure of 8.06 psi. Blowout panels protect the cabin compartments against structural damage resulting from loads developed as a result of rapid decompression. Negative pressure relief valves are provided to limit cabin negative pressure, attainable during a rapid descent following a decompression.

Cold air is produced by passing engine bleed air through the primary heat exchanger, the cooling turbine, and the secondary heat exchanger. Cooling air for the heat exchangers is provided by ram air in flight, and by heat exchanger cooling fan air on the ground. A pneumatic lockout in the form of a pressure switch in the flow control valve prevents bleed air flow in the event heat exchanger cooling is not available.

Radio rack equipment cooling is provided by exhaust air from the flight compartment. Cabin pressure forces the air through the cooling circuit and overboard through the flow limiting radio rack discharge venturi. During ground operation, and during flight upon selection by the crew, a fan forces the airflow. If the fan is required to operate and inadequate pressure rise exists across it, the radio fan off indicator light comes on.

Temperature control maintains either manually or automatically, a selected temperature in the flight or passenger compartment. The flight and passenger compartment temperature control systems are mechanically and electrically independent, and maintain temperature in one compartment regardless of the temperature in the other. Two sets of controls and indicators on the forward overhead switch panel provide automatic or manual control and monitoring of each system.

ELECTRICAL POWER SYSTEM

The AC and DC electrical power systems used on the airplane are similar in design. Referred to as a split-bus system, both the AC and DC electrical power systems are divided into two independent systems; the left side and the right side. Normally each side operates independently from the other, each having a power source and bus system supplying power to the various load demands throughout the airplane. In the event of a power loss on the load buses of either side, a crosstie relay is provided so that the dead buses can be connected to the buses of the opposite side. Two major differences between the AC system and the DC system are: (1) sensing and control relays are installed in the AC system to prevent any two power sources from operating in parallel to supply power to the AC load buses, and (2) if an AC generator bus should lose power, the AC crosstie relay is operated automatically to tie the two generator buses together if the AC bus crosstie switch is in the automatic position. When the airplane is on the ground, the control circuitry for the AC crosstie relay is inter-locked with the ground control relays to prevent crosstying when the auxiliary power unit (APU) generator or external power is supplying power to the airplane. The DC crosstie relay is actuated to tie the two DC buses together only when the DC bus crosstie switch is manually placed in the closed position.

Two engine-driven AC generators, one on each engine, are normally the primary source of power supplied to all AC load buses. Each generator is a brushless, air-cooled type, rated at 40 kva with a 120/208-volt, 3-phase, 400-cycle per second output. Generator speed is held at a constant 6000 rpm, with any variation of engine speeds between idle and takeoff, by a corresponding hydro-mechanical type constant-speed drive (CSD) transmission. Frequency is maintained at 400 (±4) cps by a governor inside the CSD.

Auxiliary AC electrical power is provided by a generator of the same type as the engine-driven generators to supply AC power to the corresponding dead buses if an engine-driven generator fails during flight, or to supply power to the AC load buses when the airplane is on the ground and external power is not available. The generator is driven by a constant-speed, gas turbine auxiliary power unit (APU) installed in the aft accessory compartment of the airplane. Through the use of control switches and relays, auxiliary power can be supplied to all AC load buses at the same time, or to only the left or right AC buses, or ground service AC bus, provided the bus selected is not receiving power from an engine-driven generator. If external power is supplying power to the bus selected, the APU generator will take over, and external power will be disconnected from the bus.

A 410-va, solid-state inverter is installed in the forward accessory compartment (see Figure 2-26) to provide 115-volt, single-phase, AC power to fulfill certain operational requirements. The inverter supplies AC power for ground refueling when only battery power is available. The inverter also supplies power to the AC emergency bus when the emergency power switch is placed in the on position. DC power necessary for operation of the inverter is taken from the battery direct bus.

The 28-volt DC electrical power is normally supplied by four 50-ampere, unregulated transformer-rectifiers (TR), installed in the forward accessory compartment. These units are powered from a 115-volt, 3-phase, AC power source. The 28-volt outputs from two of the TR's are connected in parallel to the left DC bus. The output of the third TR is connected directly to the right DC bus. The output of the fourth TR is connected to one of two buses as determined by electrical power requirements. During. During normal operation, with the right AC bus powered, the DC ground service tie relay is energized. With the relay energized, output of the TR is supplied to the right DC bus. When power for only ground servicing is required, the right AC bus is not powered, the DC ground service tie relay is deenergized, and the rectifier output is supplied to the DC transfer bus.

A reverse current relay is connected in series with the output of each TR unit as an extra precaution in the event of an internal failure of the TR which would allow DC current to flow back into the unit. If a reverse current relay is tripped, the relay must be manually reset.

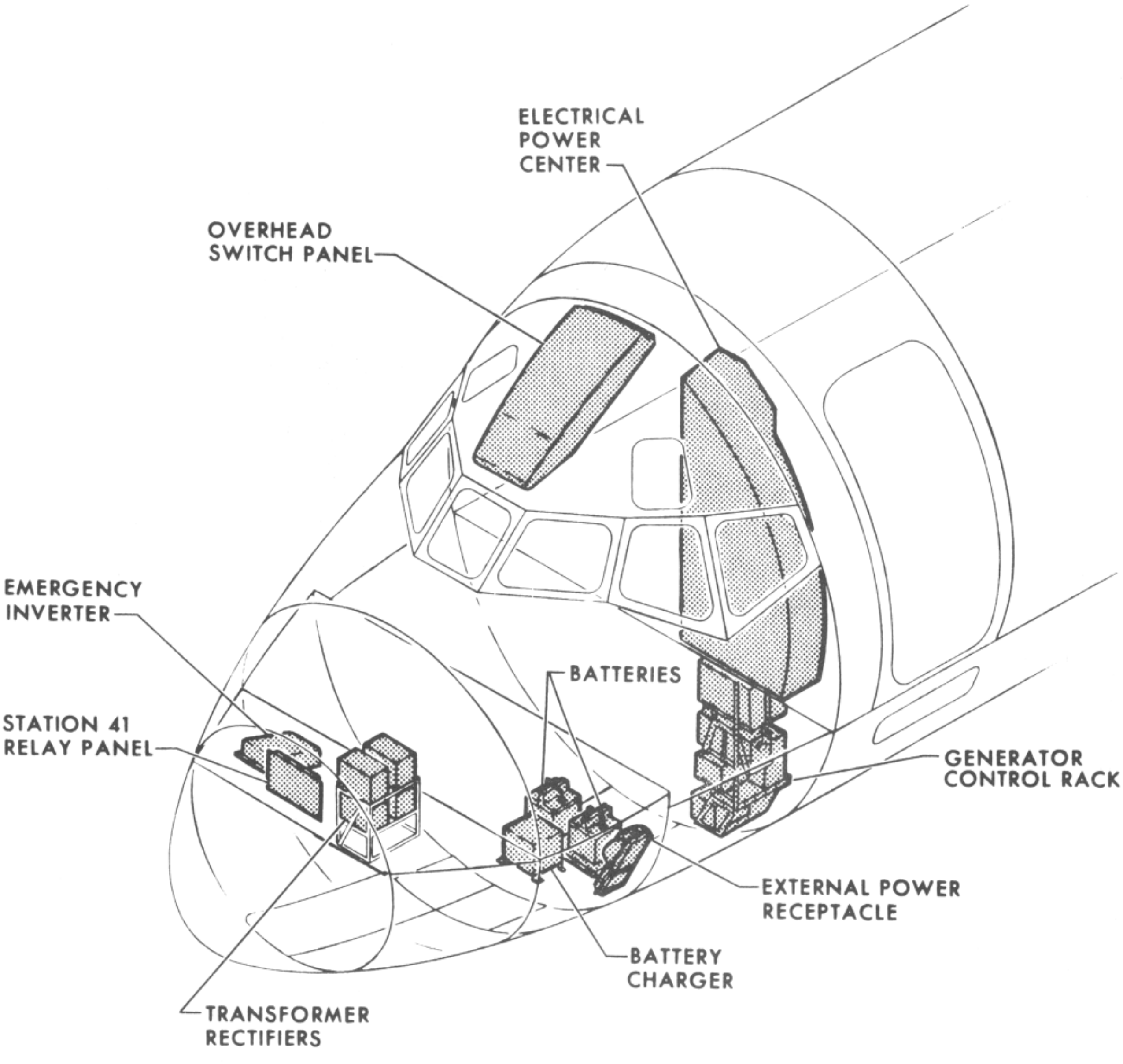

FIGURE 2-26. ELECTRICAL POWER SYSTEM COMPONENTS LOCATION

Two 14-volt batteries, installed in the electrical/electronics compartment, are connected in series to provide 28-volt DC power to the starter control relay of the auxiliary power unit and through an 80-ampere circuit breaker directly to the battery direct bus. The battery direct bus circuit breaker, located in the electrical/electronics compartment, can be manually reset from the flight compartment. A hinged handle is provided on the lower portion of the control pedestal shroud, cable-connected to the circuit breaker reset actuator. Battery power can be utilized to start the APU or for refueling operations when no other electrical power source is available. Also with a proper air supply connected to the engine starter, the batteries can supply the necessary electrical power for starting an engine. If AC power is not available during flight, the batteries are capable of supplying DC power to operate the sing-phase inverter, and to the most important DC operated equipment.

OXYGEN SYSTEM

There are two independent gaseous oxygen systems installed in the airplane, one in the flight compartment for the crew, and one in the passenger compartment for the passengers and cabin attendants (see Figure 2-27).

Each system has a thermal expansion safety discharge feature and uses a common discharge indicator. The indicator is mounted in the fuselage skin below the first officer's side window. The indicator contains a green plastic disc. Absence of the disc indicates crew or passenger oxygen cylinder pressures have exceeded maximum safety pressure.

The crew oxygen system provides the crew with oxygen to maintain a normal sustained flight, if cabin decompression should occur. In addition, the system protects the crew from harmful effects of smoke and gases. Oronasal masks are used with the system and each mask contains an integral microphone. The masks can be used with the portable oxygen cylinder.

The passenger oxygen system is designed to automatically open the mask container doors, present the masks, and supply oxygen to all emergency supplemental outlets if the cabin pressure drops to the equivalent of 11,500 ± 1000 feet altitude. The supply lines are not normally charged with oxygen. The system

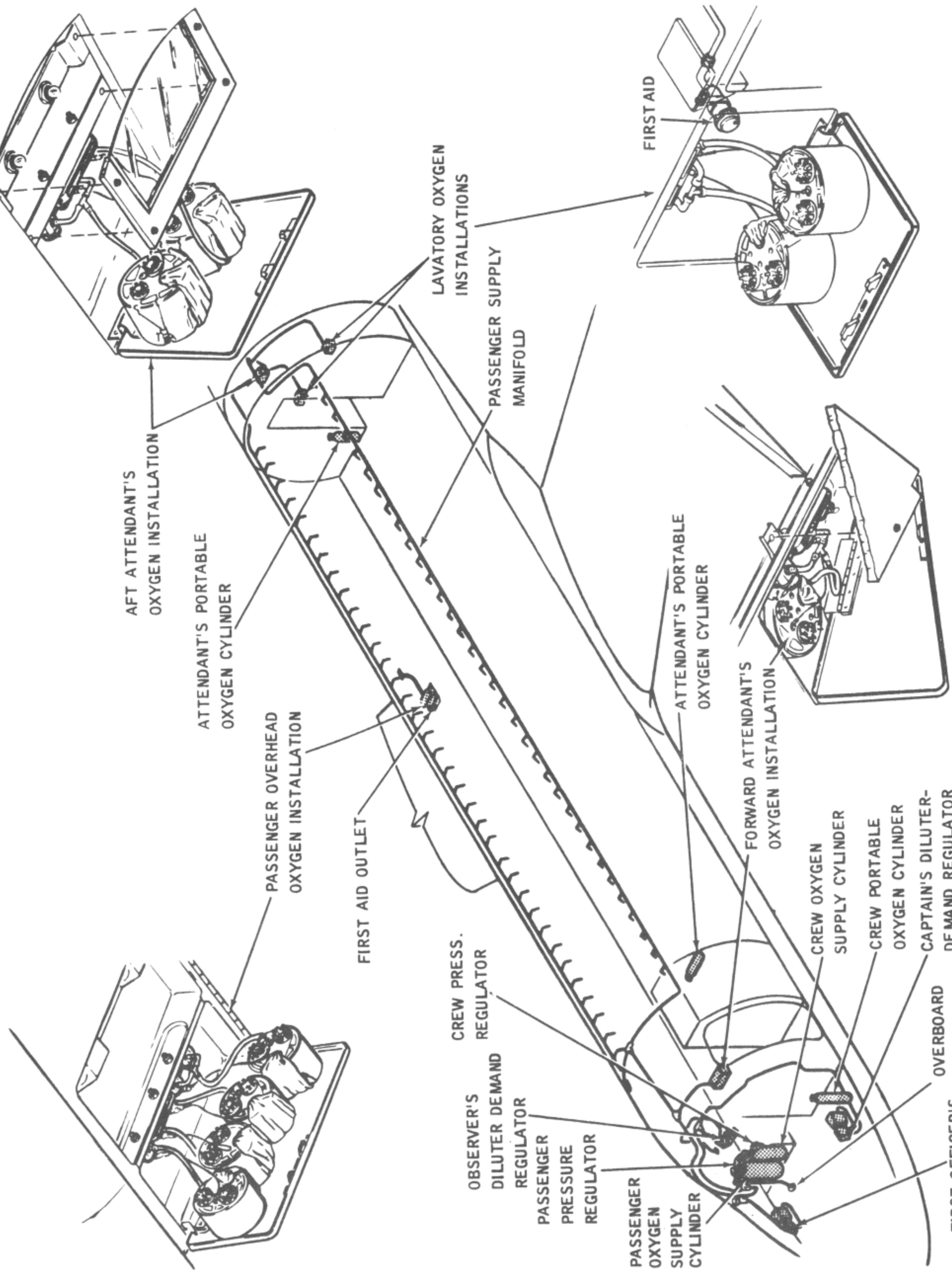

FIGURE 2-27. OXYGEN SYSTEM SCHEMATIC

can also be manually actuated on or off by the crew. On descent, an interlock prevents actuating the system automatically.

First aid oxygen in the passenger compartment is obtained from outlets located under each overhead stowage rack. In the lavatories, the first aid outlet is located inside the oxygen stowage box. Masks for first aid oxygen are located on the partition in the left coatroom below the attendant's portable oxygen cylinder.

Portable oxygen is provided for the crew and cabin attendants by individual easy-to-carry-oxygen cylinders.

3 SAFE BOMB LOCATION

This section describes the best locations to place a small bomb so that detonation will not cause catastrophic loss of the airplane. Critical equipment is identified and located with relation to the bomb.

The bomb is assumed to be equal to 4 pounds (8 sticks) of 40 percent nitroglycerine based commercial dynamite. This size bomb would probably cause major damage and every effort must be made to minimize its effect. The cruise speed should be greatly reduced to minimize dynamic loading and pilot controlled maneuvers must be restricted.

The cabin should be depressurized to prevent explosive decompression. The bomb fuse may be pressure sensitive. If the fuse mechanism is not known, descent and depressurization should be accomplished by changing the cabin pressure as little as possible. Normally, sea level cabin pressure is maintained up to 18,000 feet altitude and then decreased to 8000 feet equivalent pressure as the airplane climbs to 35,000 feet. On descent, sea level cabin pressure is regained again at 18,000 feet altitude. Altitude should be reduced to correspond to the current cabin pressure as close as possible when depressurizing. Constant cabin pressure can be maintained two ways: 1) by setting the automatic air conditioning controls to indicate a landing at the cabin pressure equivalent altitude, or 2) by manually controlling the cabin air dump valve while monitoring the cabin pressure indicator. The air conditioning system is capable of maintaining constant cabin pressure for any descent rate up to and including emergency dive speed. Even if the bomb is known to be not pressure sensitive, the altitude should be reduced to about 8000 feet if the bomb is kept in the cabin because the oxygen system may be destroyed. The oxygen system should be shut off when descending to the lower altitude for depressurization to reduce the fire hazard.

The bomb must be located as far away as possible from all systems required to land the airplane, in the region of low structural loads and isolated from the passengers and crew. Three locations are reviewed in this study: 1) in the cabin, 2) in the tail cone and 3) on the ventral stairs extended down into the slipstream.

SAFE BOMB LOCATION IN CABIN

All of the control cables, electrical lines and fuel lines are located below the floorline except for the horizontal tail trim electrical wires which are located in the tunnel overhead. (See Figures 3-1 and 3-2.) The fuel tanks are all aft of station 510 and the fuel lines extend from the tanks to the engines. The main electrical power center is located directly behind the pilot (see Figure 3-3). The emergency oxygen lines for the passengers are located in the overhead stowage rack. The oxygen lines are normally isolated from the oxygen source by a pressure valve and do not present a fire hazard below 11,500 (±1000) feet altitude. If the bomb is allowed to be detonated somewhere in the cabin compartment, it must be located away from the floor, somewhere along the length of the fuselage where the bending loads are low and away from the fuel tanks, fuel lines and electrical power center. In addition, it must be away from the passengers and crew.

Without cabin pressure, bending is the most significant load that the fuselage shell must support. Bending moments are low at the forward and aft ends and peak to a maximum over the wing. Bending loads are carried by the skin and longeron system. The skin and longerons are generally sized to meet the loads requirements. The structural material is reduced as the bending moments drop off moving away from the wing. However, the skin and longerons reach a minimum gage where no further reduction is allowed. The forward fuselage shell is designed primarily for fatigue hoop stresses due to pressure and is greatly overstrength for longitudinal bending. Minimum margins of safety calculations are a good indication of the fuselage capability to carry bending loads. The stress analyses have been reviewed for a typical configuration (DC9-31)[1*] and the margins of safety for ultimate bending are shown in Table 3-1 for the top and bottom fuselage panels. It is apparent from Table 3-1 that the forward section of the fuselage has the greatest residual strength.

The crew must be protected from the shock waves to avoid eardrum damage or worse. The threshold of eardrum damage is about 5 psi (about 16 feet for this size bomb). Therefore, the best place to put the bomb in the cabin is in the left hand hat rack about 20 feet behind the pilot

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* Numbers refer to References.

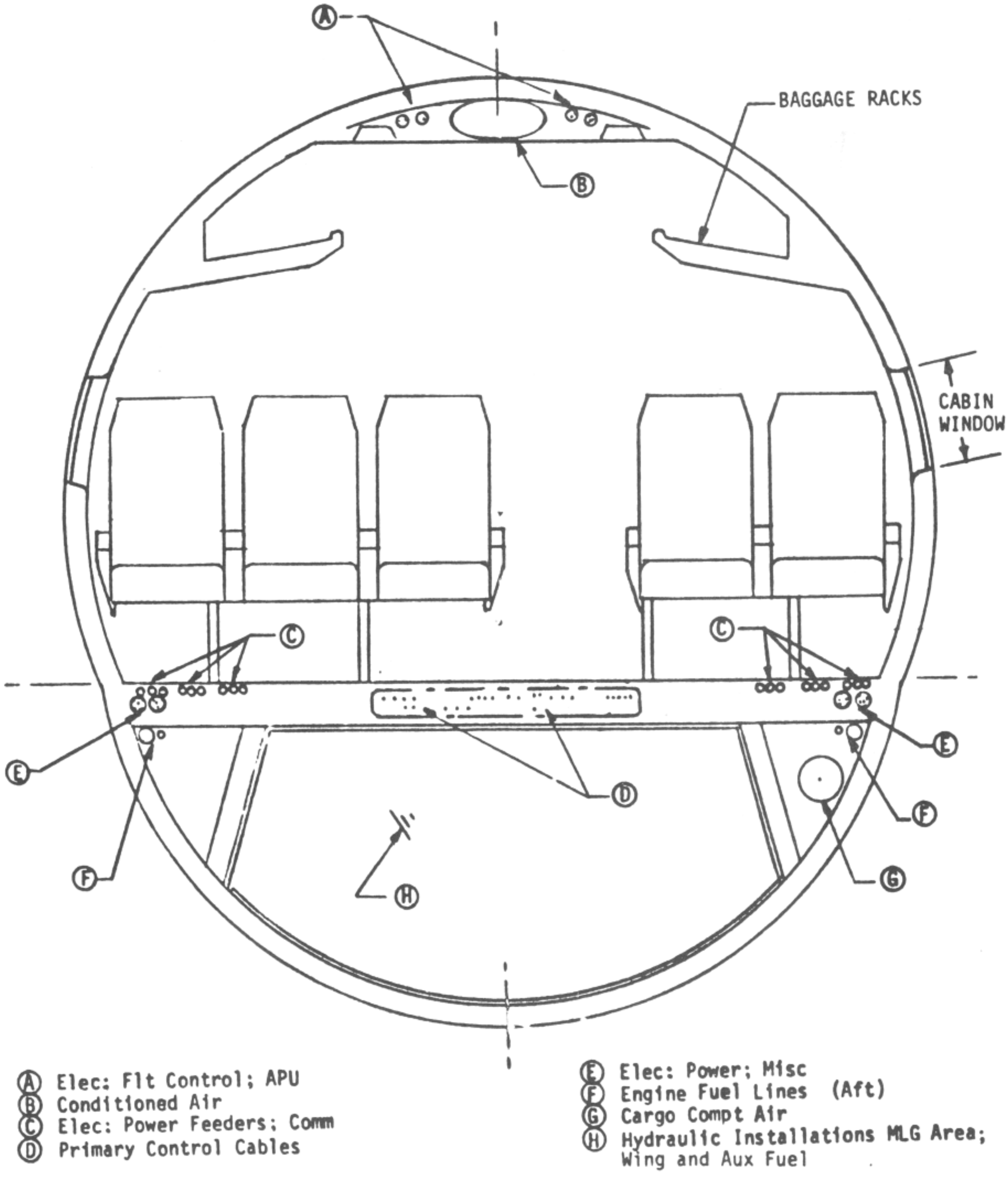


FIGURE 3-1. DC-9 FUS CABIN TYP SECTION ARRANGEMENT
(FWD OF AFT PRESS. BLKD)

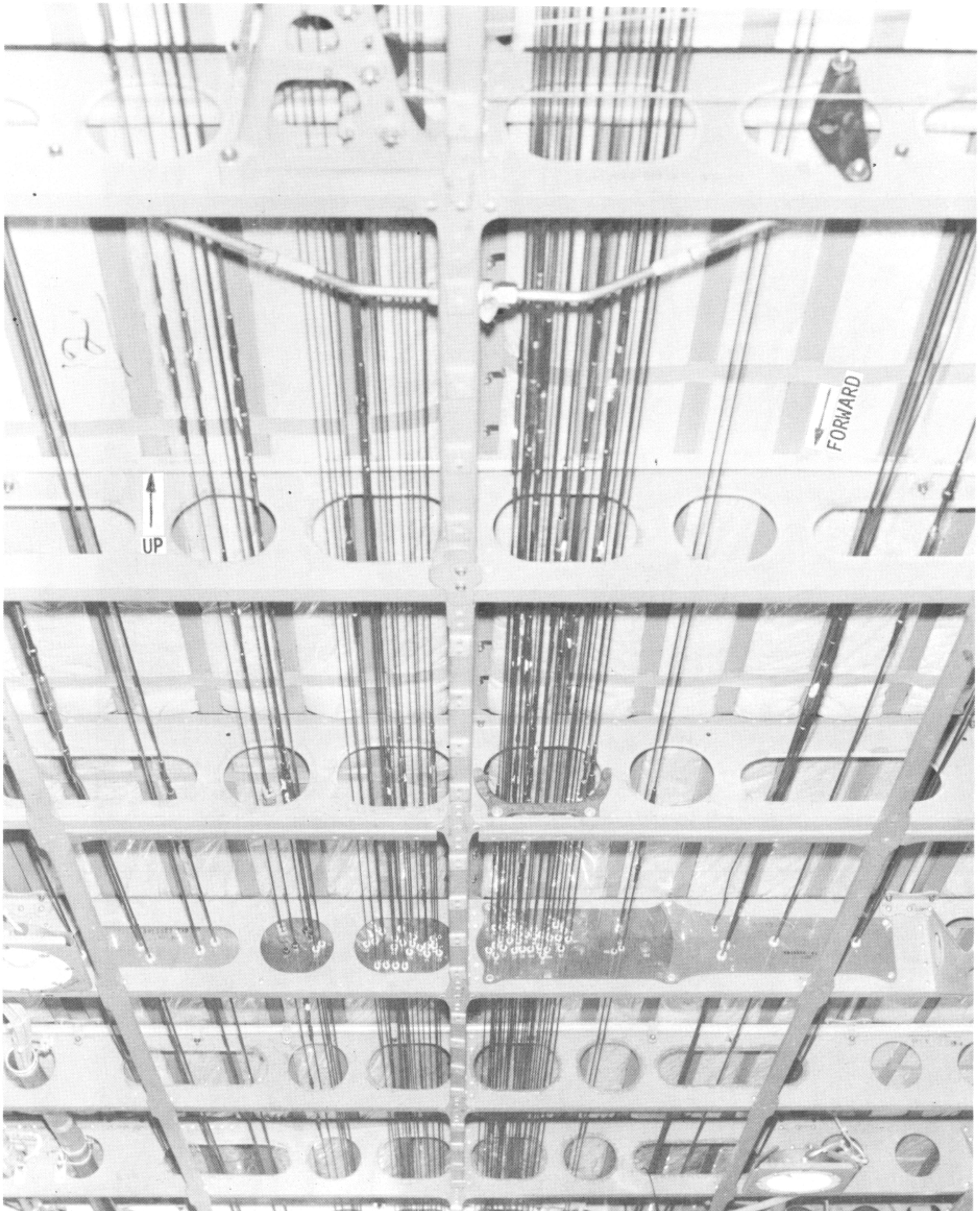


FIGURE 3-2. BOTTOM VIEW OF FLOOR IN FORWARD CARGO COMPARTMENT

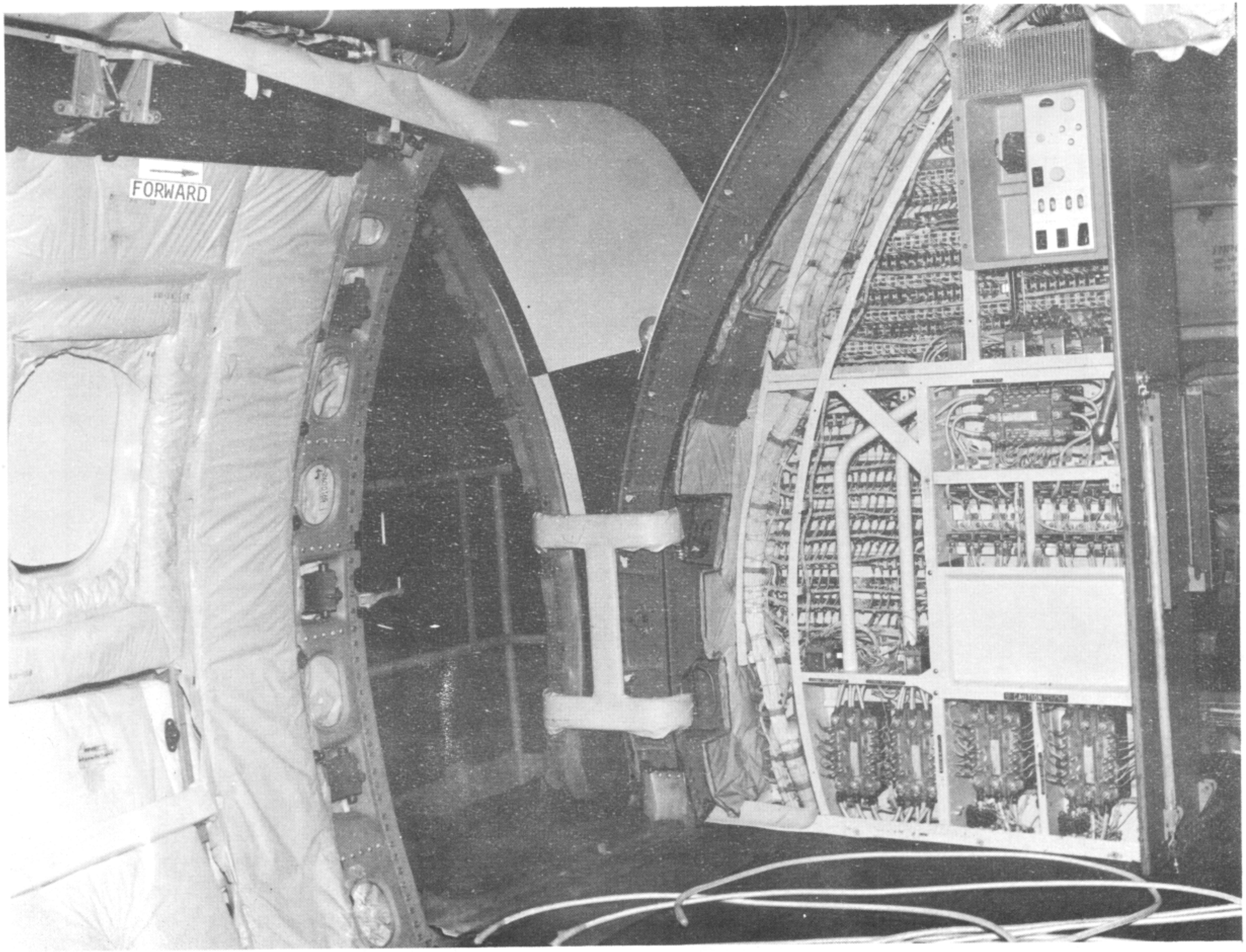
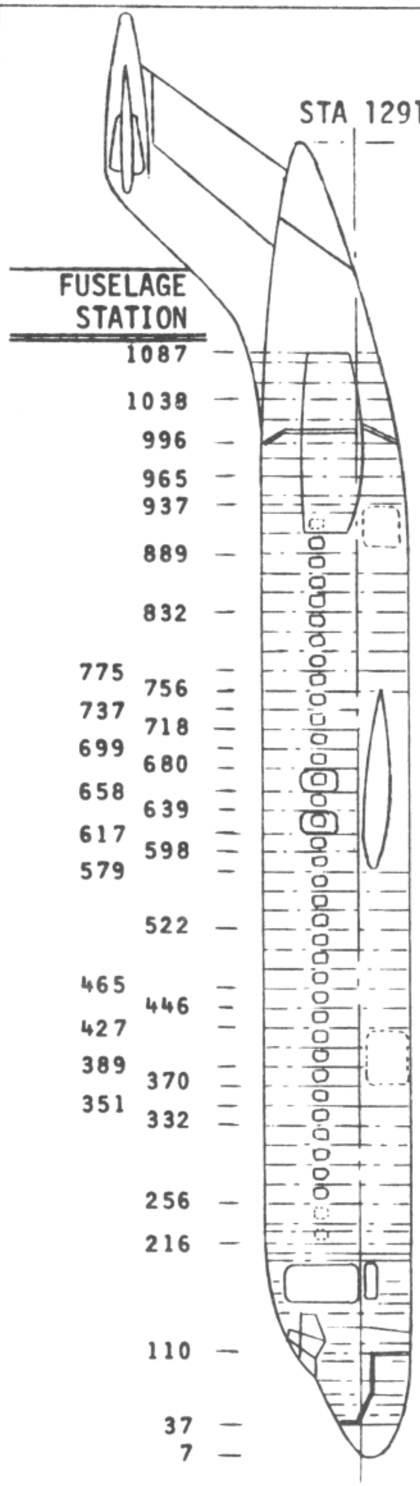


FIGURE 3-3. REAR VIEW OF ELECTRICAL POWER CENTER

TABLE 3-1. MINIMUM MARGINS OF SAFETY AT TOP AND BOTTOM PANELS FOR ULTIMATE LOAD CONDITIONS (DC-9-31)

MINIMUM MARGIN OF SAFETY FOR PANEL AT TOP LONGERON: COMBINED SHEAR AND COMPRESSION	MINIMUM MARGIN OF SAFETY FOR PANEL AT TOP LONGERON: COMBINED SHEAR AND TENSION	MINIMUM MARGIN OF SAFETY FOR PANEL AT BOTTOM LONGERON: COMBINED SHEAR AND COMPRESSION	MINIMUM MARGIN OF SAFETY FOR PANEL AT BOTTOM LONGERON: COMBINED SHEAR AND TENSION
.11	0	.41	.41
0	0	.41	.41
.04	.60	.78	.20
.13	.80	.30	.41
.33	.33	.36	
.32	.34	.22	
.18	.25	.32	
.14	.20	0	
.16	.01	.03	
.20	.04	.03	
.94	.06	.04	
.94	.15	.13	
1.00	.05	.36	
1.12	.03	.55	
1.12	.55	.07	
1.30	.57	>1.0	.09
.94	1.20	.04	1.0
1.0	1.45	.20	1.0
>1.0	>1.0	.27	1.0
>1.0	>1.0	.44	1.0
>1.0	>1.0	.41	1.0
>1.0	>1.0	.40	1.0
1.20	1.80	HIGH ↓	HIGH ↓
HIGH ↓	HIGH ↓		



(station 375). Left hand side should be used because the forward cargo compartment door is on the right hand side. Views looking forward and aft of this location are shown in Figures 3-4 and 3-5. Passengers should be located a like distance aft of the bomb and down below the top of the seat backs.

SAFE BOMB LOCATION IN TAIL CONE

The second candidate location is in the tail cone. The cone is easily accessible after the cabin has been depressurized. A walkway is provided from the rear pressure bulkhead to the forward edge of the cone. The cone is a non-load carrying fiberglass fairing and is not required for flight. Much of the structure in this area is non-critical. The region is almost void of primary equipment and is isolated from the passengers and crew. The bomb should be placed as far aft on the tail cone as possible. It could be easily placed two feet aft of the service door and pushed even further aft with aid of a stick, rod or fire ax. It could be restrained on the incline with blankets. Three important systems are in the vicinity; rudder controls, elevator controls and the structural load path for tail loads. Very little of the blast sphere would be directed toward the primary structure. Few explosive gas products would be available to pressurize the cavity.

The rudder is hinged to the rear spar of the vertical stabilizer at three points. The drive torque tube is attached to the base of the rudder at the hinge line and extends down into the aft fuselage section. The tab torque tube is bearing-mounted within the drive torque tube and supported at the lower end by a bracket attached to the empennage structure. A crank on the upper end of the tab torque tube is connected by pushrod to the control tab which is hinged to the rear spar of the rudder at three points. The drive sector, mounted at the base of the torque tubes, is connected by mechanical linkage to the hydraulic power package which is located near the top of the aft fuselage bulkhead. A two-way cable system connects the drive sector to the rudder pedals in the flight compartment. The pedals are interconnected by a torque tube below the flight compartment floor so that both sets of pedals operate in unison.

FIGURE 3-4. VIEW LOOKING FORWARD FROM SAFE BOMB LOCATION

FIGURE 3-5. VIEW LOOKING AFT FROM SAFE BOMB LOCATION

The throw-limiter actuator, on the forward side of the aft fuselage bulkhead, is connected by mechanical linkage to a hook which is positioned in a slot in the hydraulic power package actuating cylinder piston rod (see Figure 2-20). The system can be operated without hydraulic power. The rudder cables, hydraulic lines and mechanisms are directly in front and above the bomb location (see Figure 3-6. This system would be in jeopardy. However the airplane can land without rudder control and even without the rudder.

The elevator control system is operated by the control columns in the flight compartment. The control columns are interconnected so that the left and right side of the system work in unison. Control column movement is transmitted through the two-way cable systems to the control tab drive sectors and torque tubes in the horizontal stabilizer center section. Rotation of the drive sectors and torque tubes moves mechanical linkage to the control tabs. Movement of the linkage changes the position of the tab and affects aerodynamic force acting on the tab which moves the elevator. The two redundant cable systems are routed under the cabin floor and then split aft of the pressure bulkhead. One set travels up behind the bulkhead and across the top of the tail section and the other set is routed along the tail section lower surface. The systems are separated in the region of the engines so that if a turbine blade should penetrate the fuselage both cable systems would not be destroyed (see Figure 2-19). The two systems converge at the upper rear spar bulkhead and go up the vertical tail (see Figure 3-6. Here again the airplane can land without the elevator control using the horizontal stabilizer trim instead. The horizontal stabilizer is controlled electrically and the wiring is routed across the top of the cabin and up the front spar of the vertical tail.

Loads from the tail group are transmitted to the fuselage via five canted spar bulkheads in the tail section. They are front spar, rear spar, center spar and two intermediate spars. The system is redundant. The aft spar bulkhead is close to the bomb location. If the upper portion

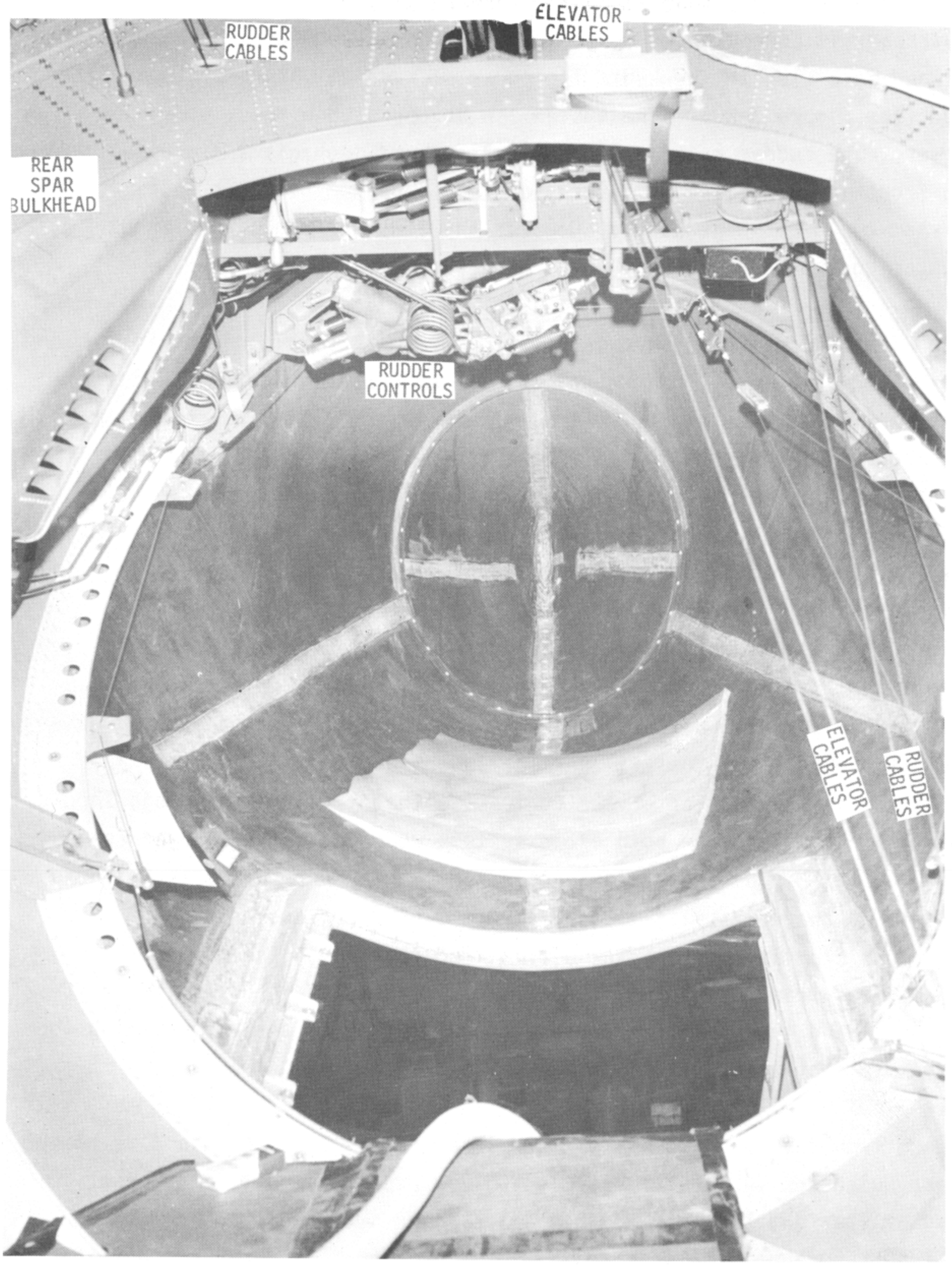


FIGURE 3-6. VIEW OF TAIL CONE LOOKING AFT

of this bulkhead and the structure forward are left intact, sufficient structure will remain to provide continuity for tail loads. The horizontal stabilizer and elevator are far enough away to survive a tail cone bomb blast of this magnitude.

SAFE BOMB LOCATION ON VENTRAL STAIRS

Ventral stairs exist on about 75 percent of the fleet. The ventral stairs are accessible after depressurizing the cabin by lifting the tail section walkway. The bomb should be placed on the aft most accessible step. The bottom step unfolds as the stairs are lowered. The stairs should be lowered into the slipstream and a crew member should walk down the stairs in order to move the folding step out of the way. The stairs will not drop very far. Calculations show that in flight the stairs can only be extended down 17 degrees with hydraulic power only and 24 degrees with the added weight of a man when the airplane is at approach speed in the full flaps landing configuration (see Figure 3-7). The bomb will be captured and held in place by the folding step when the man steps back off the stairs. A pillow or some other article may be packed around the bomb to better retain it. Placing the bomb on the bottom most step and letting the stairs retract to 17 degrees will put the bomb only about one foot outside the contour. However, as will be shown in section 4, there is an enormous benefit (reduction in damage) in getting the bomb anywhere outside the fuselage shell. This is because the impulse felt by the structure is a function of the angle of incidence; blasting 90 degrees to the surface being the worst case. The blast effect drops off drastically when the angle of incidence becomes greater than about 40 degrees. The fuselage shell would be convex to the blast and only the immediate vicinity would have a high angle of incidence when the blast is outside.

Again much of the tail structure can be destroyed and still land safely. The only requirement being that the tail group loads are transmitted through the tail stub to the main fuselage. Figure 3-8 shows the general arrangement of the tail section. Two cable systems, elevator and rudder, are routed over pulley brackets which are attached to the lower skin.

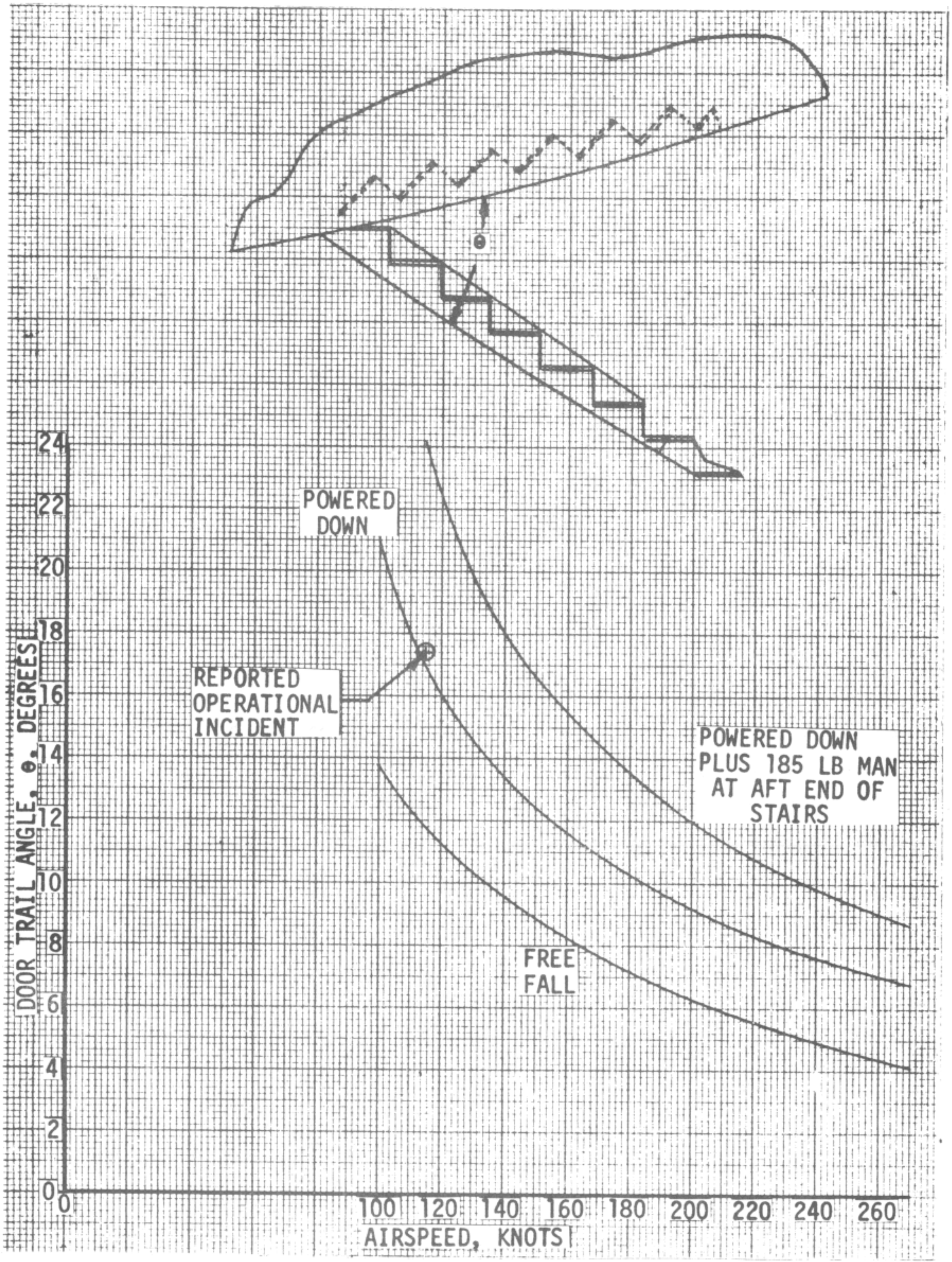


FIGURE 3-7. ESTIMATED IN-FLIGHT TRAIL ANGLE OF THE VENTRAL STAIR DOOR

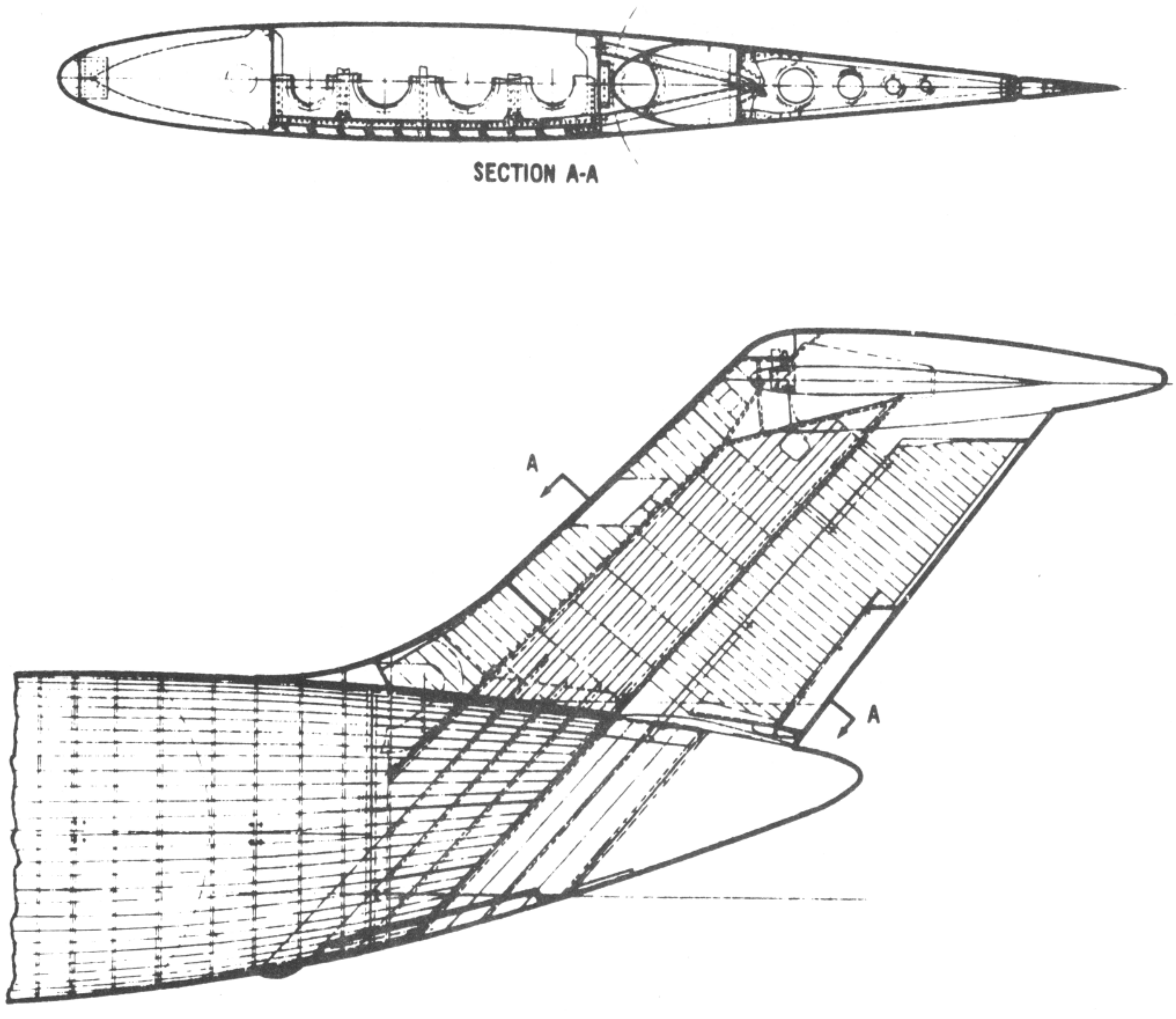


FIGURE 3-8. VERTICAL STABILIZER STRUCTURAL ARRANGEMENT

The rudder is not essential and the elevator has redundant control systems. The fuel lines, pylons and engines are forward of the stairs and a considerable distance away. The stairs are expendable. The passengers and crew are well protected forward of the pressure bulkhead. Damage in this area would least jeopardize the safety of the airplane.

4 BOMB IN THE FORWARD FUSELAGE HAT RACK

This section analyzes the damage done to the structure and calculates the strength remaining if a 4-lb. 40% dynamite bomb is detonated in the forward fuselage hat rack. The following areas are investigated:

- Free air blast effects
- Free air shock wave parameters for dynamite
- Effects of confinement
- Blast effects on structures
- Critical impact velocity
- Critical impulse in critical period
- Radial and longitudinal structural damage
- Air blast effects on passengers and crew
- Long duration pressures
- Flight load conditions
- Stress calculations for damaged structure
- Residual strength
- Summary of damage and residual strength

FREE AIR BLAST EFFECTS

When a detonation occurs adjacent to a structure such that no amplification of the initial shock wave occurs between the explosion source and the structure, then the blast loadings acting on the structure are said to be free air burst blast pressures[2] (Figure 4-1).

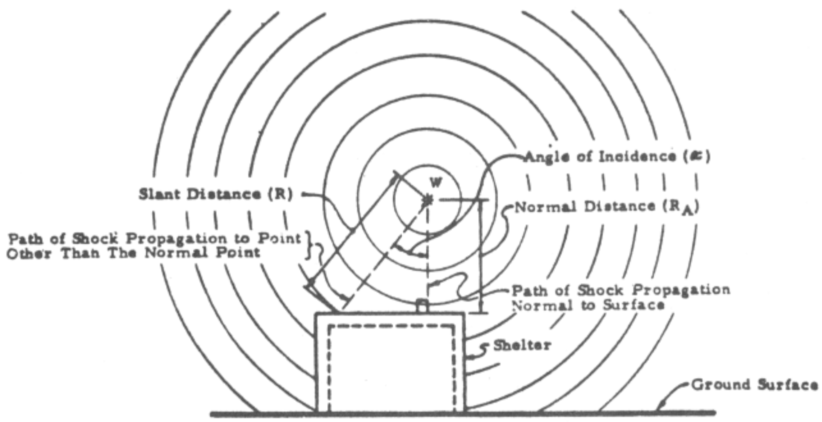


FIGURE 4-1. FREE AIR BURST BLAST ENVIRONMENT

The pressure wave appears as a sharp rise in pressure, which is the shock front, followed by a decaying portion, which is characteristic of the expansion phase of the hot gases that were produced from the explosion. At the time that the equilibrium condition essentially is established, the particles are still moving out. There is still gas flow away from the point of detonation when the pressure goes to zero, and so the system over-expands, which produces a negative pressure phase in the blast wave.

This pressure-wave curve (Figure 4-2) can be represented reasonably well by the expression[3].

$$P_s = P_{so} \left(1-\frac{t}{t_o}\right) e^{\left(\frac{-kt}{t_o}\right)}$$

where

P_{so} = Peak Side-On Overpressure (psi)

t = Time

t_o = Duration of Positive Phase

e = Base of Natural Logarithms

k = Decay Parameter

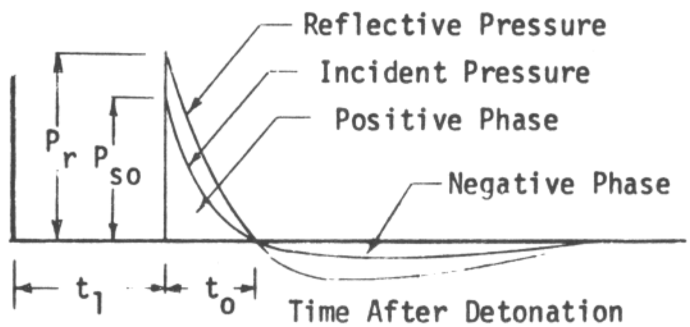


FIGURE 4-2. VARIATION OF OVERPRESSURE WITH TIME AT A GIVEN LOCATION

If k is taken as 1, the negative phase has the same total impulse as the positive phase and this is a standard assumption in treating blast waves[4]. This expression is valid only for the positive phase of the air blast and a small portion of the negative phase. The peak negative overpressure is approximately one-eighth of the peak positive overpressure, and the duration of the negative phase is approximately four times the duration of the positive phase of the air blast for low pressure.

With this expression it becomes possible to compute the impulse as a function of time for the side-on case.

The impulse as a function of time for the side-on case is

$$I_s = \int_0^t P_s \, d_t$$

This is found to be

$$I_s = P_{so} \, t \, e^{\left(-\frac{t}{t_o}\right)}$$

This expression shows that the impulse increases for a while as a function of time until the negative exponential term takes over, at which point a decrease occurs. Impulse will continue to decrease until at $t = \infty$ it will be equal to zero, which corresponds to the assumption that the impulse under the positive phase is equal to the impulse under the negative phase. Thus, at $t = \infty$ zero impulse would be imparted to the system. Since the damage is usually done long before the full effect of the negative phase comes into play, it is ignored and just the positive phase is used.

As the shock wave (incident pressure wave) moves radially away from the center of the explosion, it will come in contact with the structure. Upon contact with the structure, the initial wave pressures and impulse are reinforced and reflected (Figure 4-2). The reinforced pressures are referred to as reflective pressures, and the associated impulse is called the reflective impulse.

The basic relationships among the properties of a blast wave having a steep front are derived from the Rankine-Hugoniot[5,6] conditions based on the conservation of mass, energy, and momentum at the shock front. These conditions, together with the equation of state for air, permit the derivation of the blast wave relationships[7], presented in terms of the side-on overpressure as follows:

$$\frac{U}{C_o} = \left(1 + \frac{6}{7}\frac{P_s}{P_o}\right)^{1/2}$$

$$\frac{u}{C_o} = \frac{5}{7}\frac{P_s}{P_o}\left(1 + \frac{6}{7}\frac{P_s}{P_o}\right)^{-1/2}$$

$$\frac{\rho}{\rho_o} = \frac{7 + 6P_s/P_o}{7 + P_s/P_o}$$

$$\frac{P_r}{P_s} = 2\,\frac{7P_o + 4P_s}{7P_o + P_s}$$

$$\frac{q}{P_s} = \frac{5}{2}\,\frac{P_s}{7P_o + P_s}$$

$$\left(\frac{C}{C_o}\right)^2 = \frac{(P_s + P_o)\,(P_s + 7P_o)}{6P_s + 7P_o}$$

$$M^2 = \left(\frac{u}{C}\right)^2 = \frac{25}{7}\,\frac{P_s^{\,2}}{(P_s + P_o)\,(P_s + 7P_o)}$$

and it may be shown approximately, that

$$\frac{R}{R_o} = 0.727\ P_s/P_o$$

also

$$P_r/q = \frac{4}{5}\,(4 + 7P_o/P_s)$$

where C = speed of sound in air behind shock front

C_o = speed of sound in ambient air

M = Mach number of flow behind shock front

P_o = pressure of ambinet air

P_s = side-on overpressure

q = dynamic pressure = $1/2\rho u^2$

R = Reynolds number per foot, flow behind shock front

R_o = Reynolds number per foot for ambient air = 6.83×10^6 at sea level

u = particle velocity of flow behind shock front

U = shock-front velocity

ρ = density fo air behind shock front

ρ_o = density of ambient air

When the radius of the shock front becomes greater than the normal distance to the structure, the incident shock wave which impinges upon the structure is reflected back, forming the reflective shock wave illustrated in Figure 4-3.[8]

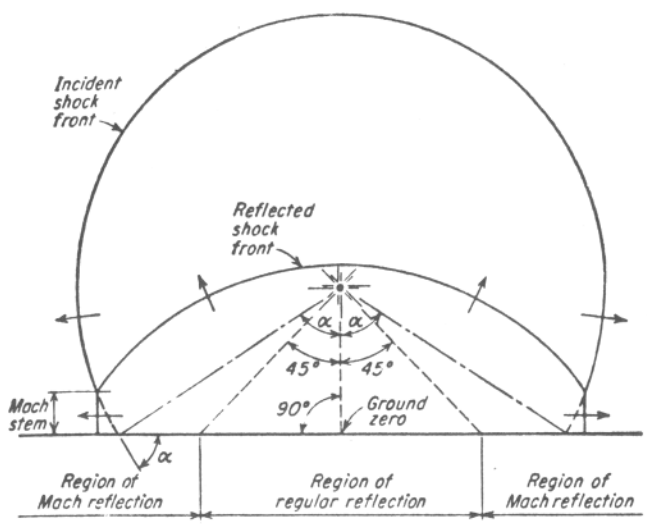


FIGURE 4-3. SHOCK-REFLECTION PHENOMENA IN REGION WHERE α IS GREATER THAN 45°

The arrows at the incident and reflected shock fronts indicate the duration in which the shock waves are traveling. The location of the reflected shock front is roughly determined by drawing an arc with the center located a distance equal to the normal distance behind the structure and directly under the point of detonation, joining the points of intersection of the incident shock with the reflecting surface. The symbol α represents the angle of incidence of the shock wave with the structure. The reflected shock-wave overpressure $P_{r-\alpha}$ is a function of the incident shock overpressure P_{so} and the angle of incidence α.

The reflected shock front in Figure 4-3 travels through the atmosphere at a higher velocity than the incident shock and gradually overtakes and merges with it to form a single shock front called Mach stem. The fused shock front thus formed is normal to and travels parallel to the structure's surface. The Mach-stem formation is initiated when the angle of incidence α of the shock wave becomes greater than approximately 45°. Once formed, the height of the Mach stem gradually increases as the radius of the shock wave becomes greater.

The region on the structure's surface within which α is less than approximately 45° and no Mach stem is present is called the region of regular reflection, while the region for which α is greater than approximately 45° and a Mach stem is present is called the region of Mach reflection. The importance of the Mach-stem phenomenon is that it causes two shock waves to fuse into a single wave of higher over-pressure and of greater destructive power to structures located in its path.

The peak overpressure P_{so} existing in the shock wave adjacent to the structure surface is a function of the distance from the point of burst and the yield of the weapon. The duration of the positive phase t_o of the blast wave is a function of the peak overpressure P_{so} and the total energy yield of the explosive. The reflective overpressure ratio $P_{r-\alpha}/P_{so}$ is plotted in Figure 4-4 as a function of the angle of incidence α of the shock front.[8] This figure applies to both an inclined shock front striking the surface of the structure and a vertical shock front striking a vertical surface at an angle of incidence α.

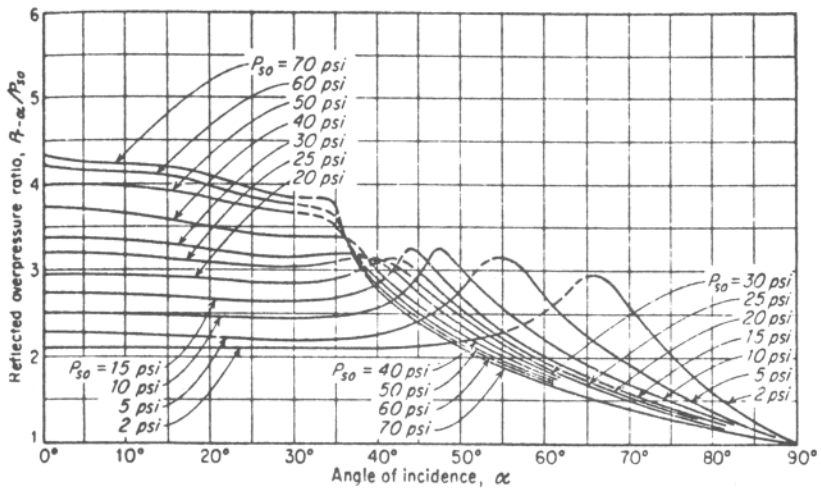


FIGURE 4-4. REFLECTIVE OVERPRESSURE RATIO VS ANGLE OF INCIDENCE FOR VARIOUS PEAK OVERPRESSURES

It has been found that air-blast phenomena such as the pressure and duration at different distance are related for different strength bombs according to the ratio of the cube root of the equivalent weights of TNT. These relationships are referred to as the scaling laws. This cube root scaling is a standard procedure and can be applied to peak pressure, impulse, and positive-phase duration. These relationships may be written in the form

$$P = f(R/W)^{1/3}$$

$$R_2/R_1 = (W_2/W_1)^{1/3}$$

$$I_2/I_1 \quad = \quad R_2/R_1 \quad = \quad (W_2/W_1)^{1/3}$$

$$t_2/t_1 \quad = \quad R_2/R_1 \quad = (W_2/W_1)^{1/3}$$

where

R = distance to point of detonation
P = overpressure
I = impulse
t = period of positive phase

The basic shock wave parameters based on test data with a reference TNT explosive are reported in Reference 2 and are shown in Table 4-1.

TABLE 4-1. SHOCK WAVE PARAMETER FOR SPHERICAL TNT EXPLOSION IN FREE AIR AT SEA LEVEL (DENSITY, 95 LB/CU/FT)

Z	P_{so}	P_{so}^-	$\frac{I_s}{W^{1/3}}$	$\frac{I_s^-}{W^{1/3}}$	U	V	$\frac{L}{W^{1/3}}$	$\frac{L^-}{W^{1/3}}$	P_r	P_r^-	$\frac{I_r}{W^{1/3}}$	$\frac{I_r^-}{W^{1/3}}$	$\frac{t_1}{W^{1/3}}$	$\frac{t_o}{W^{1/3}}$	$\frac{t_o^-}{W^{1/3}}$
$\frac{ft.}{lb^{1/3}}$	psi		$\frac{lb \quad ms}{in^2 - lb^{1/3}}$		$\frac{ft.}{sec.}$		$\frac{ft.}{lb^{1/3}}$		psi		$\frac{lb\text{-}ms}{in^2\text{-}lb^{1/3}}$		$\frac{ms}{lb^{1/3}}$		
0.136	6,900	14.70	294.0	54.20	22,500	18,532	0.136	-	82,500	14.70	4,350	54.20	-	.0384	6.240
0.177	5,880	14.70	192.0	54.20	19,600	17,154	0.177	-	67,600	14.70	3,620	54.20	.0019	.0302	6.260
0.252	4,640	14.70	91.00	54.20	16,490	15,088	0.252	-	50,000	14.70	2,600	54.20	.0062	.0400	6.350
0.379	3,240	14.70	34.00	54.20	13,450	12,595	0.379	-	31,500	14.70	1,585	54.20	.0146	.0423	6.420
0.505	2,340	14.70	22.60	54.20	11,600	10,594	0.505	-	21,450	14.70	1,080	54.20	.0239	.0462	6.560
0.757	1,320	14.30	18.20	53.00	9,400	8,003	0.505	0.252	11,420	14.50	550.0	53.00	.0476	.0538	6.740
1.262	595.0	14.00	17.00	45.50	6,750	5,445	0.430	0.782	4,650	14.40	244.0	47.50	.1130	.0770	7.110
1.770	322.0	13.30	17.00	35.00	5,250	3,877	0.430	1.332	2,240	14.00	150.0	42.80	.2040	.1150	7.450
3.525	162.0	10.30	25.90	23.70	3,700	2,650	0.530	1.990	925.0	11.80	90.00	33.80	.3910	.3000	7.800
3.780	56.00	4,400	13.70	15.20	2,400	1,440	1.010	2.070	240.0	7.050	51.00	23.80	.8500	.7400	8.050
5.050	26.80	2.800	11.50	11.50	1,880	882.0	1.385	3.650	87.50	4.700	34.00	18.50	1.500	1.112	8.200
7.570	10.90	1.700	7.900	7.700	1,475	456.0	1.940	5.620	28.20	2.940	20.00	11.00	3.120	1.690	8.300
12.62	4.100	0.960	5.000	4.500	1,275	203.0	2.620	9.100	9.400	1.760	10.80	8.300	7.000	2.385	8.400
17.70	2.600	0.560	3.600	3.200	1,300	134.0	3.000	9.140	5.700	1.310	7.500	6.200	11.23	2.785	8.450
25.25	1.700	0.480	2.600	2.400	1,170	85.00	3.330	9.260	3.500	0.940	5.200	4.500	17.85	3.080	8.550
50.50	0.710	0.270	1.400	1.200	1,150	36.00	4.125	9.260	1.440	0.520	2.700	2.500	40.50	3.700	8.550
126.2	0.260	0.110	0.540	0.500	1,140	11.80	5.125	9.260	0.470	0.220	1.100	1.080	108.8	4.710	8.550
252.5	0.100	0.060	0.270	0.270	1,135	5.600	5.740	9.260	0.210	0.110	0.500	0.540	224.0	5.300	8.550
505.0	0.040	0.030	0.120	0.120	1,130	1.600	6.200	9.260	0.090	0.050	0.200	0.230	455.0	5.720	8.550

TABLE 4-1 SHOCK WAVE PARAMETER FOR SPHERICAL TNT EXPLOSION IN FREE AIR AT SEA LEVEL (DENSITY, 95 LB/CU/FT) (CONTINUED)

P_{so} = Peak Positive Incident Overpressure, psi

$P_{\bar{so}}$ = Peak Negative Incident Overpressure, psi

P_r = Peak Positive Normal Reflective Overpressure, psi

$P_{\bar{r}}$ = Peak Negative Normal Reflective Overpressure, psi

$I_s/W^{1/3}$ = Scaled Positive Incident Impulse, ln-ms/in^2-lb$^{1/3}$

$I_{\bar{s}}/W^{1/3}$ = Scaled Negative Incident Impulse, lb-ms/in^2-lb$^{1/3}$

$I_r/W^{1/3}$ = Scaled Positive Normal Reflective Impulse, lb-ms/in^2-lb$^{1/3}$

$I_{\bar{r}}/W^{1/3}$ = Scaled Negative Normal Reflective Impulse, lb-ms/in^2-lb$^{1/3}$

$t_1/W^{1/3}$ = Scaled Time of Arrival of Blast Wave, ms/lb$^{1/3}$

$t_o/W^{1/3}$ = Scaled Duration of Positive Phase, ms/lb$^{1/3}$

$t_{\bar{o}}/W^{1/3}$ = Scaled Duration of Negative Phase, ms/lb$^{1/3}$

$L/W^{1/3}$ = Scaled Wave Length of Positive Phase, ft/lb$^{1/3}$

$\bar{L}/W^{1/3}$ = Scaled Wave Length of Negative Phase, ft/lb$^{1/3}$

U = Scaled Shock Front Velocity, ft/sec

W = Charge Weight, lb

R = Radial Distance from Charge, ft

Z = Scaled Distance, $R/W^{1/3}$, ft/lb$^{1/3}$

FREE AIR SHOCK WAVE PARAMETER FOR DYNAMITE

Commercial blasting explosives, with the exception of black powder, are referred to as dynamites, although in some cases they contain no nitroglycerin. The standard 40% dynamite used by the U. S. Bureau of Mines in its comparative tests contains nitroglycerin 40%, sodium nitrate 44%, wood pulp 15% and calcium carbonate 1%. Other suitably modified compositions make 20% or 60% dynamites. Still further compositions are classified by an indication of the standard composition with approximately the same performance; e.g., 40% dynamite defines any composition which has approximately the same performance as the standard 40% dynamite, although it may contain a different percentage of nitric esters and a different active base. The strengths of several of these dynamites relative to trinitrotoluene (TNT) are shown in Table 4-2.[9]

TABLE 4-2. EQUIVALENT WEIGHTS

		EQUIVALENT WEIGHT PERCENT TNT
STRAIGHT DYNAMITE	40%	65
	50%	79
	60%	83
AMMONIA DYNAMITE	40%	41
	50%	46
	60%	53
GELATIN DYNAMITE	40%	42
	50%	47
	60%	76

The shock wave parameters for 4 pounds of 40 percent nitroglycerin based commercial dynamite are derived from the TNT reference data from Table 4-1 using the relationship in Table 4-2 that the strength of 40% dynamite is equal to 65% TNT. Then per the scaling laws

$$(W_{DYN})^{1/3} = (.65W_{TNT})^{1/3}$$

where

W_{DYN} = WEIGHT OF DNYAMITE

W_{TNT} = WEIGHT OF TNT

The results are shown in Figure 4-5.

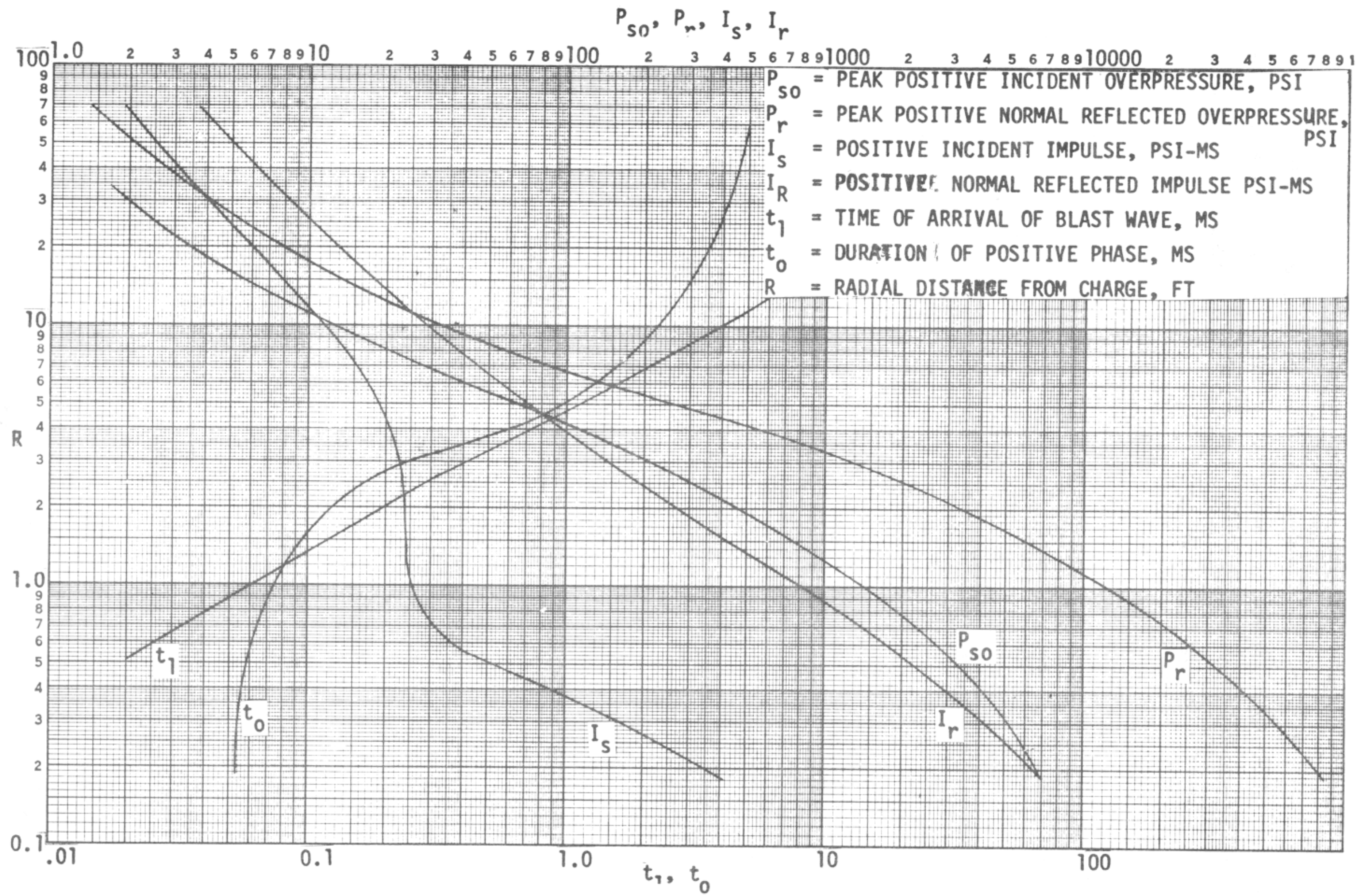


FIGURE 4-5. SHOCK WAVE PARAMETERS FOR A 4-LB SPHERICAL 40-PERCENT STRAIGHT DYNAMITE EXPLOSION IN FREE AIR AT SEA LEVEL

EFFECTS OF CONFINEMENT

When an explosion occurs within a structure, the peak pressures associated with the initial blast output (free air pressures) will be extremely high. These pressures in turn will be amplified due to their reflections within the structure. In addition, the accumulation of gases from the explosion will exert additional pressures and increased durations within the structure. The combined effects of both pressures may eventually destroy the structure unless venting for the gas and the shock pressures is provided. If some of the structure is sufficiently frangible, adequate venting can occur. To minimize the reinforcement of pressures of the blast output, the frangible section must fail vertually at the time of onset of the initial blast loads. This failure will relieve the amplified pressures and minimize the reinforcement and reflection to other non-frangible elements of the structure. As may be seen in Figure 4-5, the impulse and pressures are extremely high for a radius of one foot and less. Therefore airplane shell is frangible for at least a two-foot diameter section adjacent to the bomb.

BLAST EFFECTS ON STRUCTURES

The overpressure-time history of a shock wave has little correspondence with the load experienced by a structure enveloped by the shock wave.[6] When a blast wave encounters an obstacle, the wave is diffracted by the obstacle. The pressure loadings developed by the reflected shock waves during this "diffraction phase" of the loading history are considerably higher than the static side-on overpressure; by a factor of two for weak shocks, the multiplying factor for pressure increasing to twelve[2] for strong shocks. Despite those very high pressures, the diffraction phase is often not of great structural significance, since the duration of the diffraction loading is characteristically short. For very-high-frequency structures, however, diffraction loading is the predominant cause of damage.

After the blast wave has engulfed the structure, the pressure differential between the front and back faces of the structure drops to a relatively low value. The structure is being compressed by the static pressure of the blast wave, and the loading in the direction of motion of the blast is a function of the dynamic pressure or stagnation pressure of the flow behind the shock front. This is the "drag phase" of the loading history - much less intense than the diffraction phase but of much longer duration, hence often the main factor in producing structural damage.

Blast targets are classified as high-frequency, or "diffraction" targets; or low-frequency, or "drag" targets. At first glance the classification would be assumed to imply that the diffraction targets are damaged by the diffraction phase of the loading and that the drag targets are affected by the drag phase. The former part of the statement is fairly accurate; the latter much less accurate, since there are low-frequency targets which fail during the drag phase which have had their primary impetus toward failure provided by the impulsive (relative to their natural periods) velocity change of the diffraction phase.

A more precise way of thinking of this classification would be to consider diffraction targets those failing during the diffraction phase of the loading and the drag targets those failing during the drag phase.

DRAG TARGET $t_o \leq \frac{1}{4} T_n$

DIFFRACTION TARGET $t_o > 0.8\ T_n$

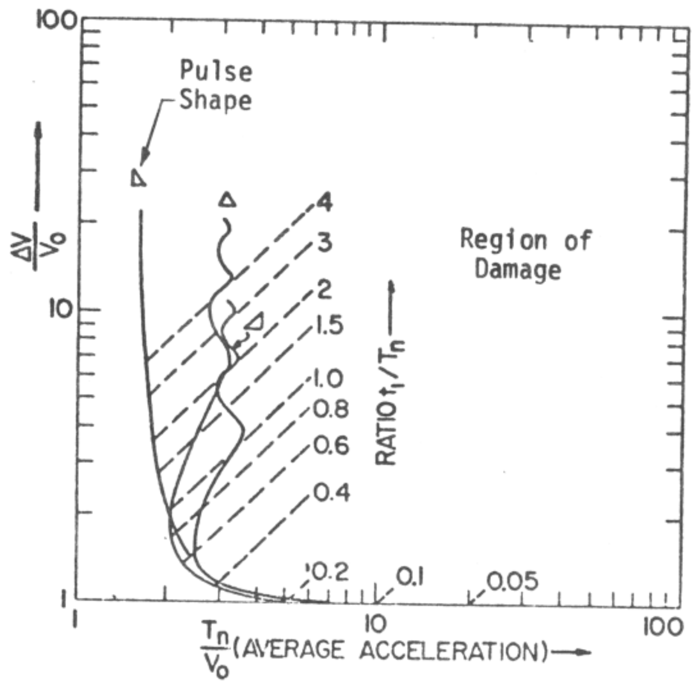
t_o = DURATION OF IMPACT

T_n = NATURAL PERIOD OF STRUCTURE

Low-frequency structures such as the fuselage shell may be analyzed rather easily if they meet the short-duration-loading criteria (drag target) in which case the criterion of damage from impact is velocity change, ΔV.[6]

CRITICAL IMPACT VELOCITY

When a single degree mass-spring system is subjected to a triangular impulse the specimen will fail or not according to a sensitivity curve as shown in Figure 4-6.[6] If the velocity change (ΔV) and average acceleration of the impulse lie below or to the right of the curves, the specimen will experience no damage; damaging pulses fall above and to the right of the curves.



V_o = velocity of mass as it passes through the neutral position sufficient to produce a specified travel

ΔV = integral of acceleration

T_N = natural period of impact

T_1 = duration of impact

average acceleration = $\Delta V/T_1$

FIGURE 4-6. SENSITIVITY CURVES FOR MASS-SPRING SYSTEM SUBJECTED TO TRIANGULAR PULSES

Figure 4-6 indicates that the sensitivity curve for an identical set of test specimens is really a wide band in the long-duration region. Obviously, the location of the vertical (or long duration) asymptote is a function of the shape of the input pulse. This implies the necessity of knowing more about the input pulse than ΔV and average acceleration. Note, however, that the horizontal (or short-duration) asymptote is independent of pulse shape, and may be specified by a unique value of velocity change. This constitutes a great simplification of some impact situations, the only specification being that the duration is short enough to be on the short-duration asymptote.

"Zones"of impact" are assigned to the impact sensitivity curve:

1. Short duration impact zone.

$$0 < \frac{t_1}{T_n} < 0.4, \Delta V = V_o$$

   In this zone, onset rate, pulse shape, or peak acceleration have negligible influences; only velocity change is important.

2. Intermediate-duration impact zone, $0.4 < t_1/T_n < 3$, $1 < \Delta v/V_o < 10$. Both onset rate and total pulse duration are important, with a spread of a factor of two in acceleration required to cause a given degree of damage at a fixed value of velocity change.

3. Long-duration acceleration zone, $t_1/T_n > 3, \Delta v/V_o > 10$. Onset rate is the primary influence here, with a spread of a factor of two between the slowly rising acceleration pulse and the infinite-onset-rate pulse.

Complex systems behave as simple mass-spring systems if the following requirements are met:

1. Failure is characterized by the amount of strain which is attained by elastic action.

2. Loading of the structure is uniform, and the dynamic deflection of the structure has the same mode shape as that of the fundamental natural frequency.

Critical impact velocities for aluminum alloys are shown in Table 4-3 (References 10 and 11).

TABLE 4-3. CRITICAL IMPACT VELOCITIES OF ALUMINUM ALLOYS

METAL	CONDITION	CRITICAL IMPACT VELOCITY (FT/SEC)	
		EXPERIMENTAL	THEORETICAL*
2S	ANNEALED	>200	176
2S	1/2 HARD	110	36
24S	ANNEALED	>200	174
24ST	AS-RECEIVED	>200 & 238	290

* Computed from static engineering stress-strain curve

CRITICAL IMPULSE IN CRITICAL PERIOD

The blast damage criterion of "critical impulse in critical time" has been used in weapon effectiveness studies[3] and is used in this analysis to determine extent of probable damage.

When a structure such as a simple sheet is exposed to a blast wave it will begin to move as the pressure acts on it. An assumption may be made that there exists a critical working velocity of the elements of the plate relative to fixed points on the plate; and that if the pressure wave incident on the target is capable of producing a velocity greater than this critical working velocity, then permanent deformation will result. Whereas the critical working velocity is not known for most materials, the critical impact velocity is known. This velocity is associated with tensile failure in the material and has also been related by John Pearson and John Rinehart[10] to the critical particle velocity in the material: that velocity or relative velocity between the two particles in the material, which if exceeded causes the particles to behave as independent particles and no longer as part of the whole system. Critical impact velocity is used in this anlaysis. In addition, this critical velocity must be achieved in one-quarter of the period the fundamental resonant frequency of the structure.

The critical impulse per unit area is:

$$I_c = \rho t V_c \qquad \text{(psi-ms)}$$

where

ρ = density of structure = .101 lb/in^3 in for aluminum

t = thickness of structure

V_c = critical velocity

The limits of probable damage can be expressed:

$$\int_o^{T_N/4} P_s \, dt \geq I_c \qquad \text{sure damage}$$

$$\int_o^{T_N/4} P_{r-\alpha} \, dt \leq I_c \qquad \text{no damage}$$

where

P_s = side-on pressure, psi

$P_{r-\alpha}$ = reflective pressure, psi

$T_N/4$ = the quarter period, ms; when $T_N/4 > 1$, the integration is taken from o to T_o

t = time

T_o = duration of positive phase of wave

The resonant frequency for rectangular plates is[12]:

$$f = \frac{C\ t\ k}{a^2} \qquad \text{(CPS)}$$

where

C = frequency constant varies with plate aspect of ratio a/b

t = plate thickness (in.)

a = length of short side (in.)

b = length of long side (in.)

k = material factor = .985 for aluminum

The natural period of the plate is:

$$T_N = \frac{1}{f}$$

EXAMPLE: Calculate critical impulse and critical time for tearing the skin of the upper panel.

$t = .050$ in., $a = 7.55$ in., $b = 19$ in.

$\rho = .101\ \text{lb/in}^3$, $V_c = 238$ ft/sec (Table 4-3)

$$I_c = \frac{.101\ \text{slug}}{386.4\ \text{in}^3} (.050\ \text{in}) \left(238 \times 12\ \frac{\text{in}}{\text{sec}}\right) 1000 = 37.3\ \text{psi-ms}$$

$$\frac{b}{a} = \frac{19}{7.55} = 2.52$$

$C = 231015$ for clamped plate (Ref. 12)

$$f = \frac{(231015)(.050)(.985)}{(7.55)^2} = 199.596\ \text{CPS}$$

$$T_N/4 = \frac{1000}{4f} = 1.2\ \text{ms}$$

The critical impulse and critical time for the upper panel, side panel and floor are shown in Table 4-4.

TABLE 4-4. CRITICAL IMPULSE AND CRITICAL TIME FOR FUSELAGE STRUCTURAL ELEMENTS

LOCATION	THICKNESS	PANEL SIZE	CRITICAL IMPULSE	FREQUENCY CONSTANT	RESONANT FREQUENCY	¼ NATURAL PERIOD
	(IN.)	(IN.)	$I_c = \rho t V_c$ (psi-ms)	(REF 12)	$f = \frac{Ctk}{a^2}$ (CPS)	$T_N/4$ (ms)
UPPER PANEL	.050	7.55x 19	37.3	231015	199.6	1.2
SIDE PANEL	.063	8.30x 19	47.0	234490	211.2	1.18
FLOOR	.025	2.5x 19	18.7	225600	888.9	0.28

RADIAL AND LONGITUDINAL STRUCTURAL DAMAGE

Figure 4-7 shows the location of the bomb in the hat rack relative to the structure. Points at one foot radial distances are noted around the shell floor. The reflective impulse that each region feels is a function of the angle of incidence and the reflective overpressure ratio. The reflective impulse for each point is calculated in Table 4-5.

The various blast data calculated thus far are collected in Figure 4-8. The reflective impulse for the particular points, incidence impulse, normal reflective impulse, and duration of the positive pulse are plotted as a function of the radial distance from the bomb blast. The critical impulse and critical time for side panel, upper panel and floor are superimposed on the same figure.

The criteria developed above states that no damage will occur if the reflective impulse is less than the critical impulse or arrives later than one-quarter of the natural period of the structure. Also damage is sure to occur if the incident impulse is greater than the critical impulse and arrives within one-quarter of the natural period of the structure. This indicates a wide range possibilities. As may be seen from Figure 4-8, the incident impulse exceeds the critical impulse for the side and upper panels only in the first half foot and for the floor in the first five feet (floor being six feet away). On the other hand, the reflective impulse is greater than the critical impulse for many of the points and critical time is the controlling element. But the limiting time of one-quarter of the natural period is arbitrary and only a rough sort of cut-off time for impulse effectiveness.

Many factors contribute to the bomb effectiveness. The interior liner and insulation will absorb some of the energy and lengthen the duration of the pulse. Increasing the duration of the given impulse results in the decrease of the achieved particle velocity and amplitude. This decrease is relatively small until the duration of the impulse becomes about

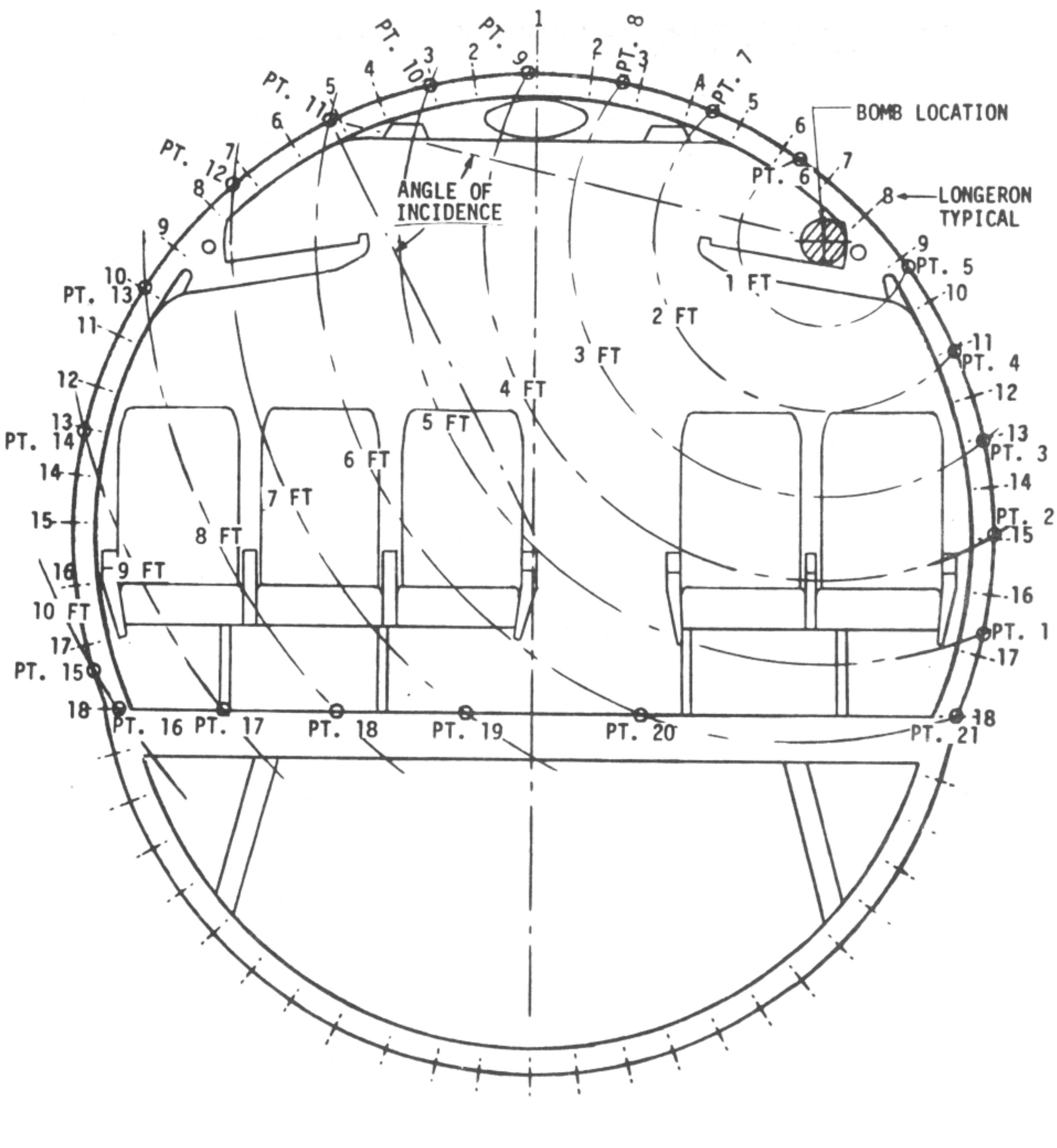
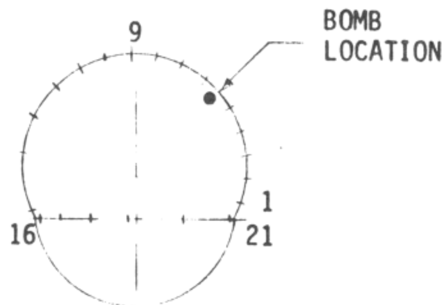


FIGURE 4-7. FORWARD FUSELAGE BOMB LOCATION

TABLE 4-5. REFLECTIVE IMPULSE CALCULATIONS FOR 4-LB 40-PERCENT DYNAMITE BLAST LOCATED IN HAT RACK

LOCATION (FIG. 4-7)	ANGLE OF INCIDENCE α DEG. (FIG. 4-7)	RADIAL DISTANCE FROM BOMB FT. (FIG. 4-7)	INCIDENT IMPULSE I_s psi-ms (FIG. 4-5)	NORMAL REFLECTIVE IMPULSE I_{r-90^o} psi-ms (FIG. 4-5)	REFLECTIVE OVERPRESSURE RATIO $P_{r-\alpha}/P_{so}$ (FIG. 4-4)	REFLECTIVE IMPULSE $I_{r-\alpha}$ $I_s(P_{r-\alpha}/P_{so})$ psi-ms (1)
1	55	5	19	74	1.8	34
2	60	4	21	100	1.7	36
3	62	3	22.5	148	1.6	36
4	62	2	23.5	270	1.6	38
5	47	1	25.5	810	2.3	59
6	47	1	25.5	810	2.3	59
7	62	2	23.5	270	1.6	38
8	62	3	22.5	148	1.6	36
9	60	4	21	100	1.7	36
10	55	5	19	74	1.8	34
11	50	6	18	57	2.2	40
12	47	7	16	46	2.4	38
13	37	8	14.5	38	3.2	38
14	28	9	13	33	3.1	33
15	15	10	11.5	29	2.7	29
16	55	10	11.5	29	2.3	26
17	51	9	13	33	2.5	33
18	45	8	14.5	38	2.8	38
19	36	7	16	46	3.3	46
20	17	6	18	57	4.0	57
21	17	6	18	57	4.0	57

(1) Limited by I_{r-90^o}



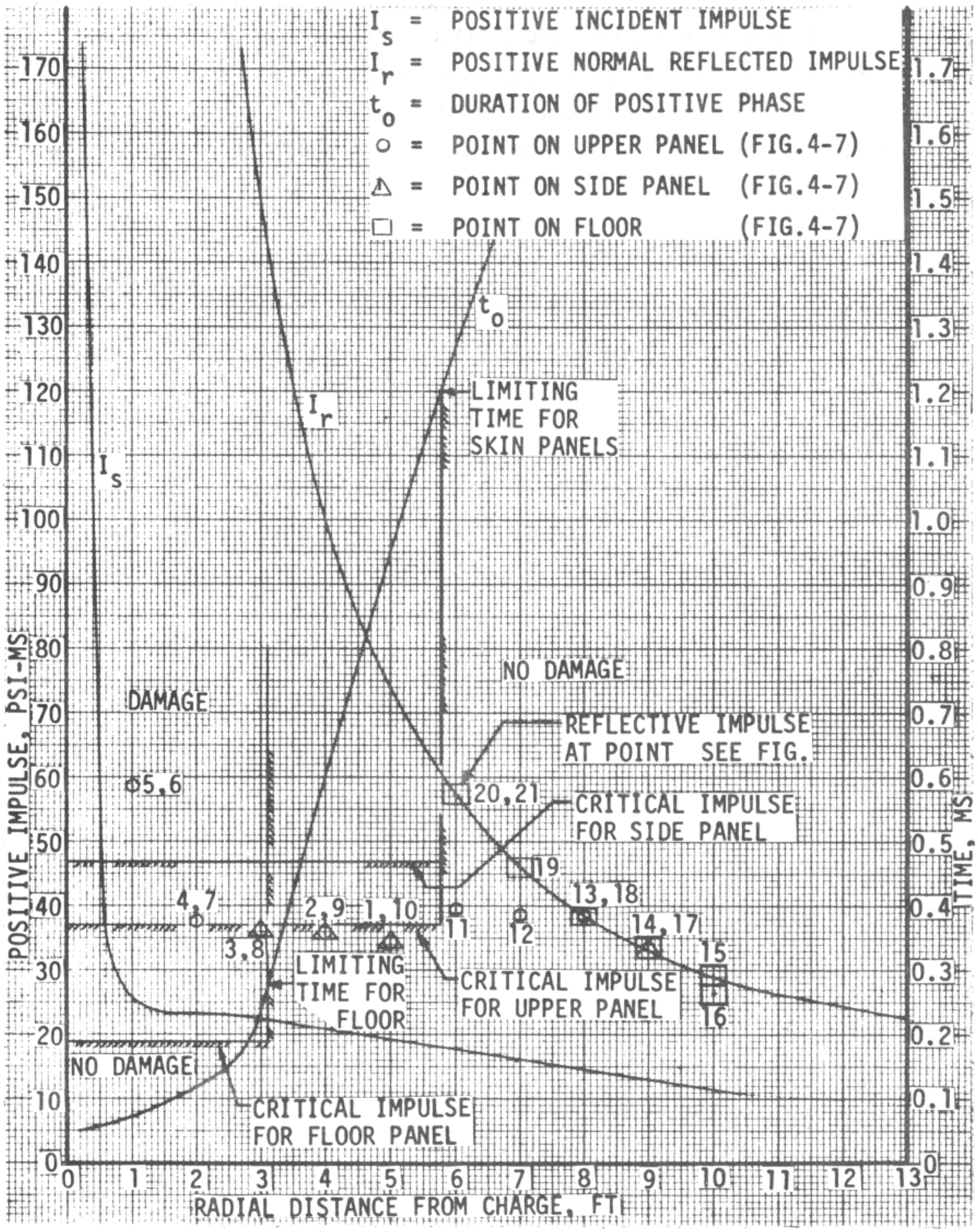


FIGURE 4-8. POSITIVE IMPULSE AT STRUCTURAL ELEMENTS OF DETONATION STATION FOR 4-LB 40 PERCENT STRAIGHT DYNAMIC EXPLOSITION IN HAT RACK

one-quarter of the natural period of oscillation, when further increases give marked decreases in amplitude[13]. The same benefit can be stated for the carpet on the floor. The shock wave parameters are based on sea level but damage will be less at altitude[14]. The shock waves will be reflected by the many interior surfaces and will reinforce the reflected pressure and add to the total impulse felt by each element. In addition the basic shock wave data must be regarded as only approximately correct. All this leads to uncertainty about the exact extent of damage. Still it seems reasonable to estimate the limit of <u>possible damage</u> according to the "critical impulse-to-critical time" criteria. Using the limits shown in Figure 4-8, the region of possible damage extends from the left hand longeron 11 across the top of the airplane to the right hand longeron 7.

The damage in the longitudinal direction is determined in the same manner. The points in the damage area at one foot radial distances (Points 4-11) are projected forward and aft at one foot intervals. The radial distances and angles of incidence increase by vector addition. New reflective impulses are calculated and the results are plotted in Figure 4-9. This figure is similar to Figure 4-8 except it is limited to the upper panel. It can be seen in Figure 4-9 that the impulse one foot forward and aft of the detonation is about the same as at the detonation station. Farther away the impulse drops off in a uniform manner. Figure 4-9 indicates that the possible damage area is rectangular extending about 4 feet in the longitudinal direction.

This limits the damage to the upper panel; about 4 feet of fuselage length from left hand longeron 11 to right hand longeron 7. In the near vicinity of the bomb where the impulse is extremely high all of the structure will be blasted away. Further away the skin will be blown off the longerons. Some of the longerons may remain intact. On the fringes of the damaged area, the skin will be torn and jagged. As will be seen later, adequate residual strength will remain to enable the airplane to land safely.

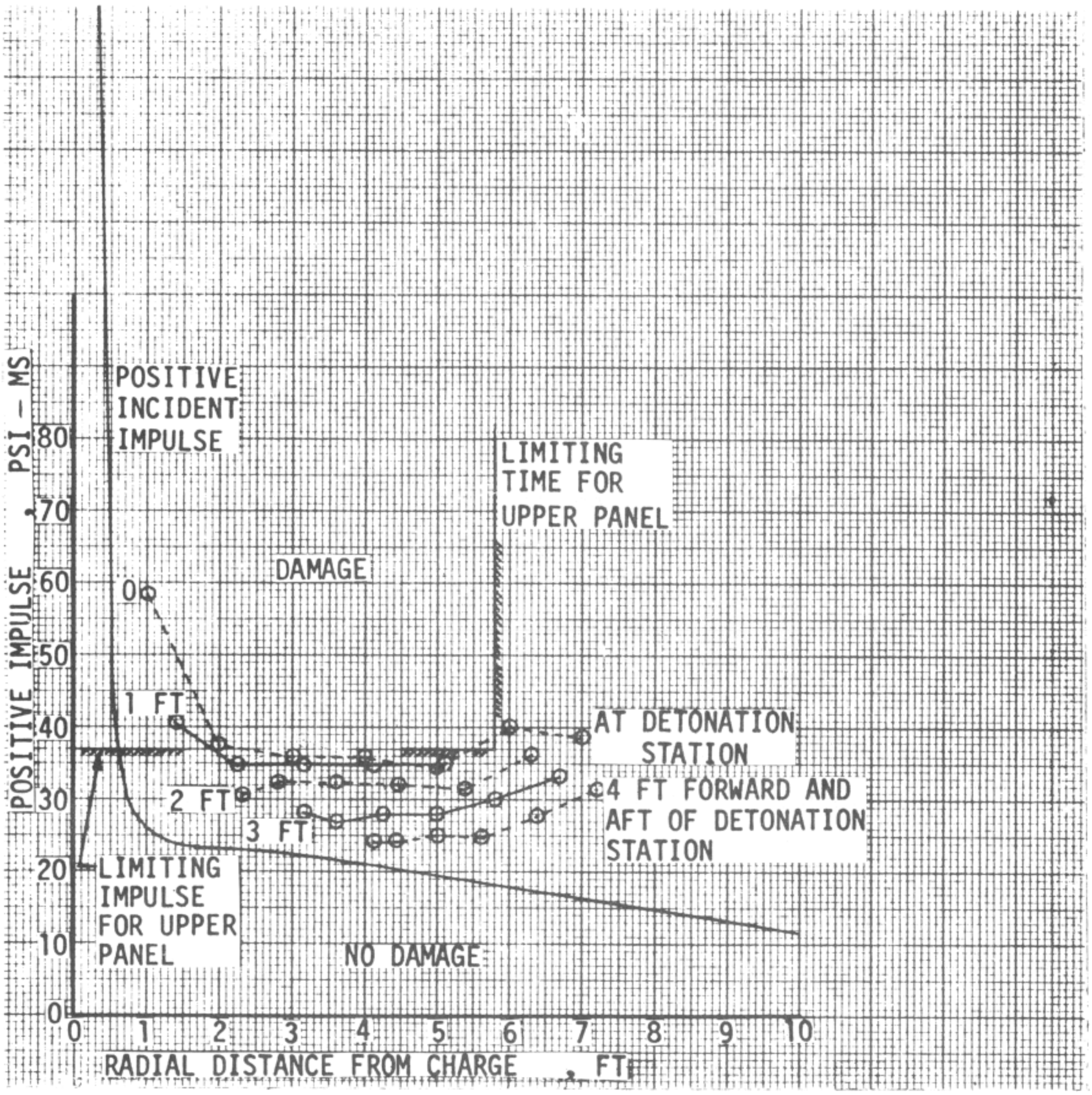


FIGURE 4-9. POSITIVE IMPULSE AT STRUCTURAL ELEMENTS FORWARD AND AFT OF DETONATION STATION FOR A 4-LB 40-PERCENT STRAIGHT DYNAMITE EXPLOSION IN HAT RACK

AIR BLAST EFFECTS ON PASSENGERS AND CREW

The estimates for maximal effective pressures that may be received from the incident, incident plus dynamic, or reflected pressure, depend chiefly on orientation. For an individual against a reflecting surface that is normal to the incident shock or prone with the charge detonated overhead, the maximal effect dose is the reflected pressure. If, however, the man is standing a few feet from this same reflecting surface or directly below the charge, he is subjected to pressures that rise in two steps: where, in the former position, the maximal effective pressure would probably be the incident plus the dynamic pressures in the first step and, in the latter, only the side-on incident pressure in the initial step. For personnel standing or prone side-on to the charge when it is detonated at or near the surface, the side-on incident plus dynamic pressures become the effective pressure; however, with orientations end-on in this situation, only the side-on incident pressure is the maximal effective pressure.[15]

Figure 4-10 shows the peak incident and peak normal reflective over-pressures as a function of distance from the center of the blast. Passengers and crew will be subjected to the incident overpressure (P_{so}) if they are protected from the direct blast wave. The cockpit partition and the passenger seat backs will provide this protection. People in direct line with the blast will feel the normal reflected overpressure (P_r). Figure 4-10 also shows the pressures and associated distances required to inflict various degrees of injury.[16] The threshold of eardrum damage is about 5 psi. The pressure is below 5 psi at all distances greater than 16 feet. Placing the bomb at station 375 (20 feet from the pilot) will provide ample distance to protect the pilot and copilot. Passengers should be moved a like distance aft of the bomb and situated completely below the top of the seat backs.

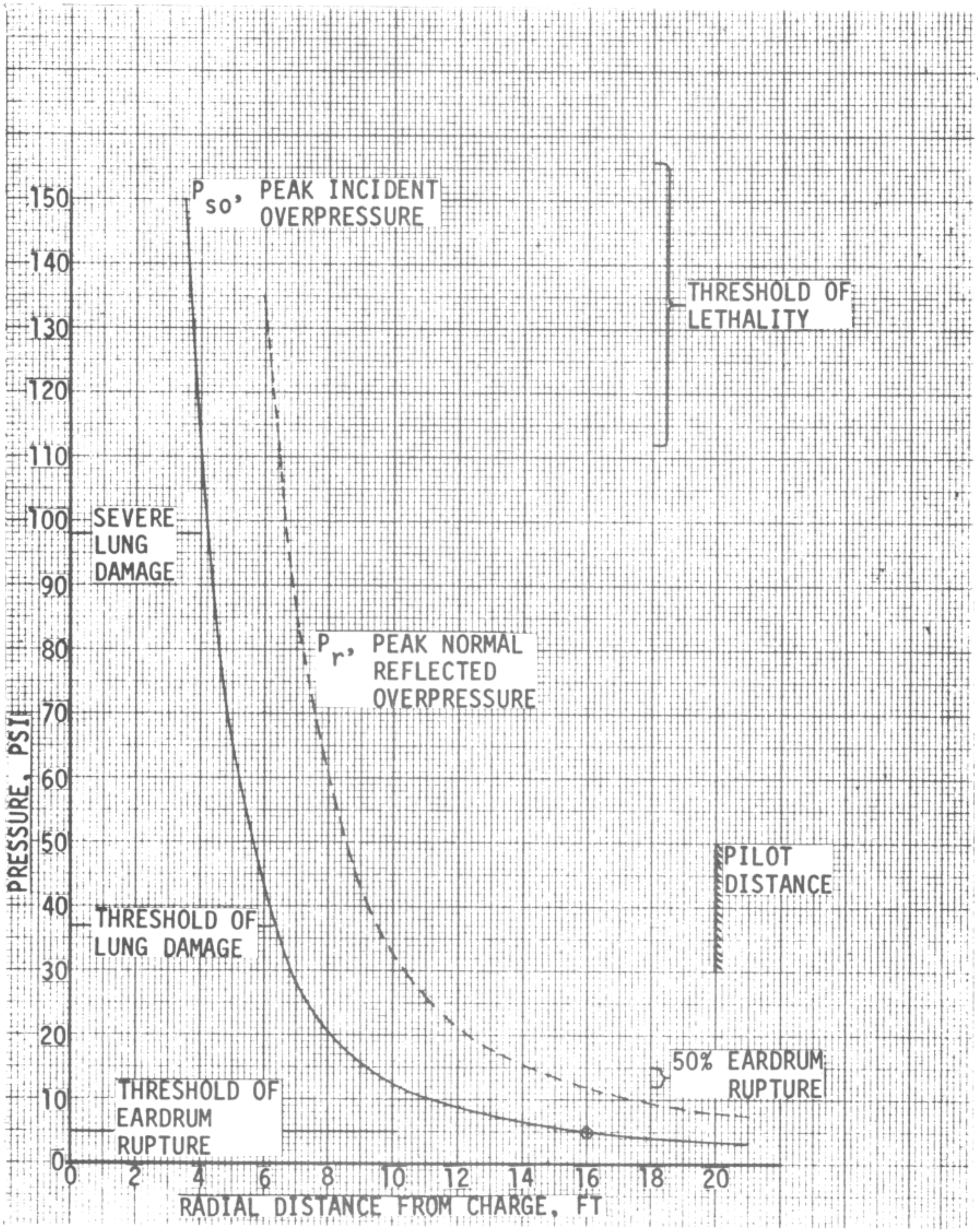


FIGURE 4-10. OVERPRESSURE VERSUS DISTANCE FROM 4-LB 40-PERCENT STRAIGHT DYNAMITE EXPLOSION

LONG DURATION PRESSURES

For a high explosive detonated in a partially closed chamber, a long duration pressure develops subsequent to the shock wave propagation. The leak area of the chamber will mainly determine the duration of the resulting blast wave and the weight of the charge will determine the peak pressure.

The pressure record shown in Figure 4-11 is an example of the pressure-time history recorded at a point on the wall of a closed or partially closed chamber when a charge is detonated in the center of the chamber. At first, the pressure shows several very high peaks, which, however, are of very short duration. These peaks are caused by repeated reflections against the walls. The fuselage shell and floor will fail or not during their short duration according to the critical impulse-to-critical time criteria. The mean pressure, denoted $P_{m(t)}$ in Figure 4-11 has a long duration compared to the natural period of the structure and can be considered as the long duration static pressure. Tests have shown that the leak areas have no noticeable influence on the peak mean pressure $P_{m(o)}$[17].

Prior to bomb detonation, the cabin is depressurized and the pressure across the floor is equalized. Upon detonation in the cabin, the region above the floor will be pressurized higher than below the floor. The fuselage shell is designed to withstand 16 psi ultimate internal pressure.[1] The floor is designed to withstand a differential ultimate pressure acting down of 1.95 psi.[18]

If the bomb is detonated in the hat rack, the near vicinity of the shell will be blasted away and the gas products in that section will be vented overboard. Referring to Figure 4-8, the region of sure damage is about one foot in diameter (points 5 and 6). This represents about 25% of the blast sphere.

Test data[17] of peak mean pressure ($P_{m(o)}$) for TNT is shown in Figure 4-12 as a function of charge-volume ratio. The volume of the cabin and cockpit is 3516 cubic feet[19]. The charge volume ratio for 4 lbs. of dynamite using 65% of TNT weight (see Table 4-2) is:

$$\frac{Q}{V} = \frac{(.65)(.75)(4\ LB)}{3516\ FT^3}\ \frac{(16.01818\ Kg/m^3)}{LB/FT^3} = .008\ Kg/m^3$$

This charge-volume ratio is very low and the test data in Figure 4-12 indicates no appreciable pressure rise. It is estimated that the peak mean pressure rise will be less than one-tenth of one atmosphere (1.47 psi) and the floor has sufficient strength to support this load.

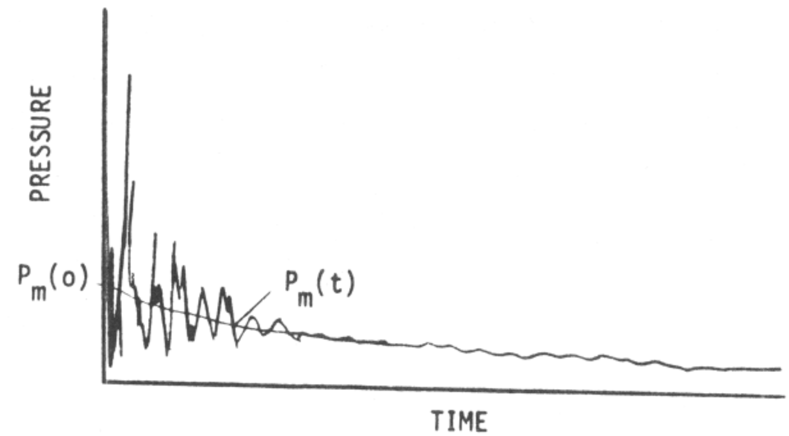


FIGURE 4-11. PRESSURE-TIME CURVE RECORDED AT A POINT ON THE WALL OF THE CHAMBER

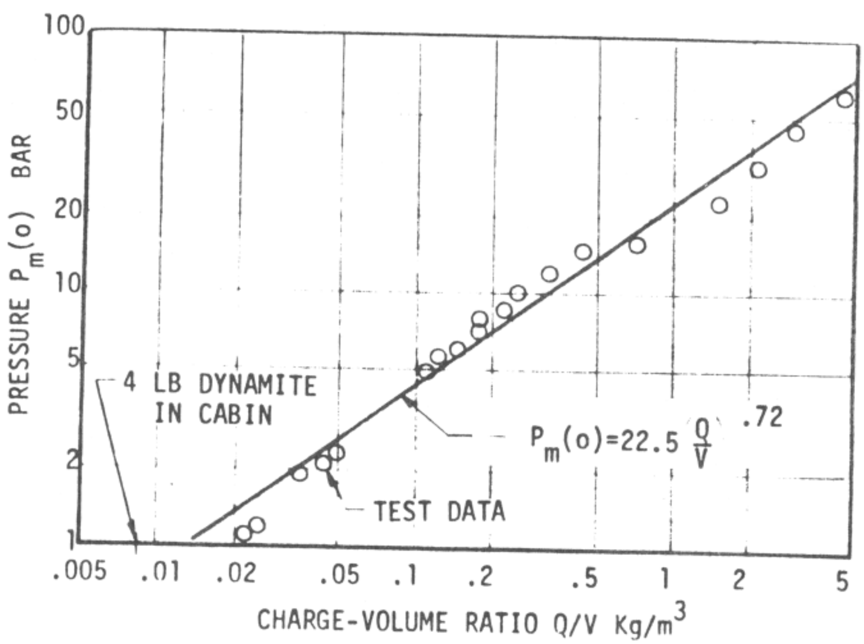


FIGURE 4-12. TEST RESULTS, PRESSURE VERSUS CHARGE-VOLUME RATIO FOR TNT

FLIGHT LOAD CONDITIONS

The fuselage is essentially a single cell thin walled beam subject to pressure, bending, torsional and axial loads. Longitudinal stiffeners are spaced about every eight inches around the periphery and frames are spaced 19 inches apart down the length of the fuselage (see Figure 2-6). The longerons and skin act together to carry the overall axial and bending loads. The frames hold the fuselage contour, limit the column length of the longerons, transfer the inertia loads to the basic shell and redistribute the shear at the discontinuities. The frames and skin carry the hoop pressure loads and the skin carries the torsional loads.

If the bomb is detonated in the forward fuselage hat rack it will completely destroy the longerons, frames and skin in the very near vicinity. A little further away the skin will be blasted off of the structure. On the fringes of the damaged area, the structure will be torn and jagged.

With the cruise speed and altitude greatly reduced and the fuselage depressurized (before detonation) the principal loading will be longitudinal tension at the damaged region and longitudinal compression at the fuselage belly. No hoop loading will exist. The frames will be intact in the belly and continue to provide column support for the compression loaded longerons. The lack of frames in the damaged area will not detract from its longitudinal tension capability. Shear will be carried by the skin and floor.

The airplane is designed to withstand loads resulting from many combinations of airspeed and load factors on the entire maneuver and gust envelopes. In order to land the airplane after massive bomb damage, action must be taken to minimize structural loading. Cruise speed must be reduced to surpress dynamic loading and maneuvers under the pilot's control must be greatly restricted. Under these conditions, the loads that the damaged section must support are very low compared to the ultimate design loads. The damaged area must be able to support the 1g loads plus a small increment to account for maneuvering and dynamic effects. Two conditions are examined in this analysis: 1) 115% of 1g vertical loading and 2) 1g vertical loading plus 10% of the maximum limit dynamic overswing lateral loading. The shears and moments for these load conditions are shown in Figures 4-13 and 4-14 along with the shears and moments used to substantiate the fail-safe strength after the failure of a single principal structural element. The shears and moments at station 400 for these two load conditions are shown in Table 4-6.[20]

TABLE 4-6. SHEARS AND MOMENTS AT STATION 400

CONDITION	VERTICAL		LATERAL		AXIAL	
	MOMENT M_y (IN-LB)	SHEAR V_z (LB)	MOMENT M_z (IN-LB)	SHEAR V_y (LB)	MOMENT M_x (IN-LB)	AXIAL (LB)
VERTICAL BENDING	-5234015	-23422			±55344	0
LATERAL BENDING	-4551317	-20367	±317313	±1272	±50839	0

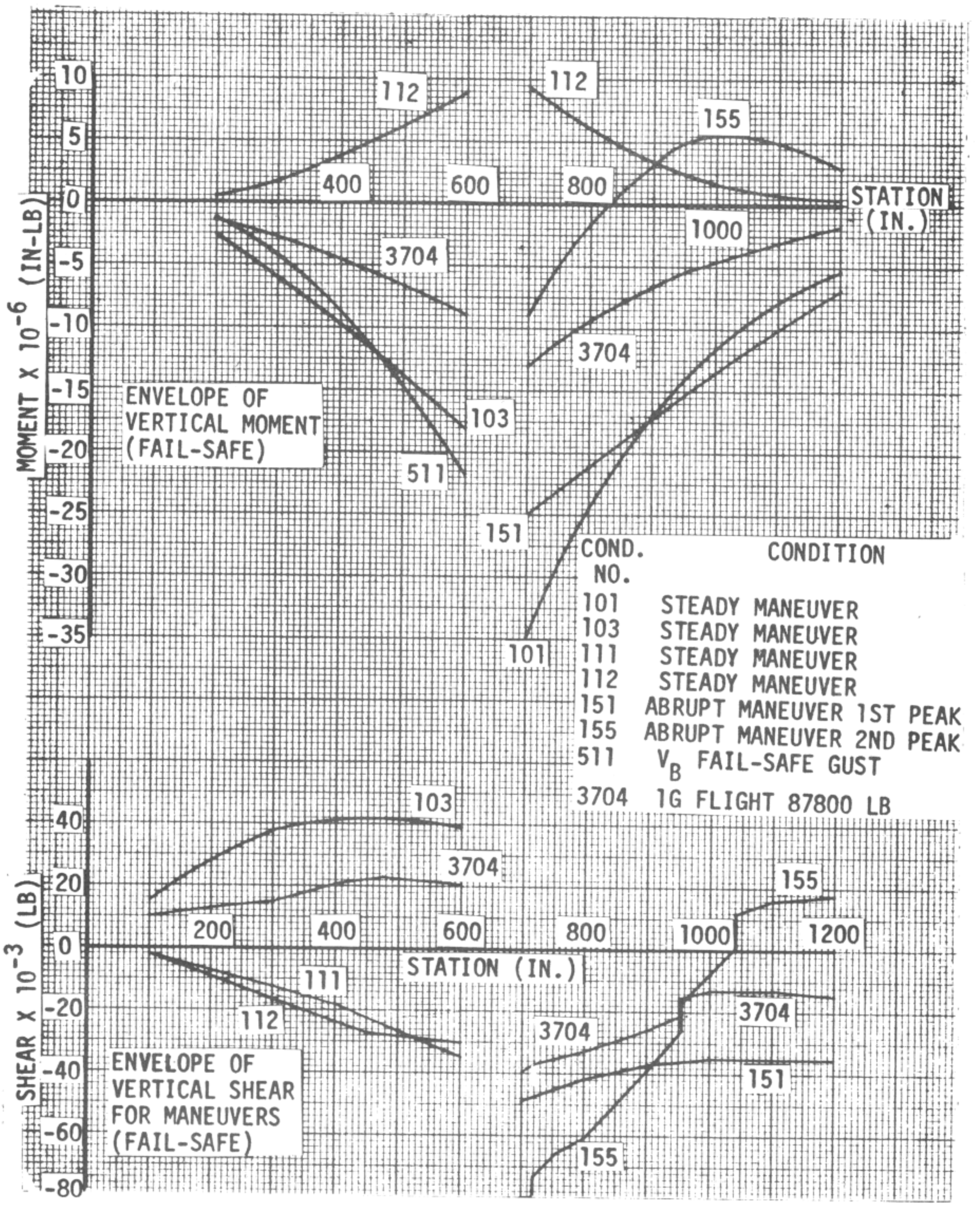


FIGURE 4-13. ENVELOPE OF VERTICAL FAIL-SAFE SHEAR AND MOMENT (DC-9-31)

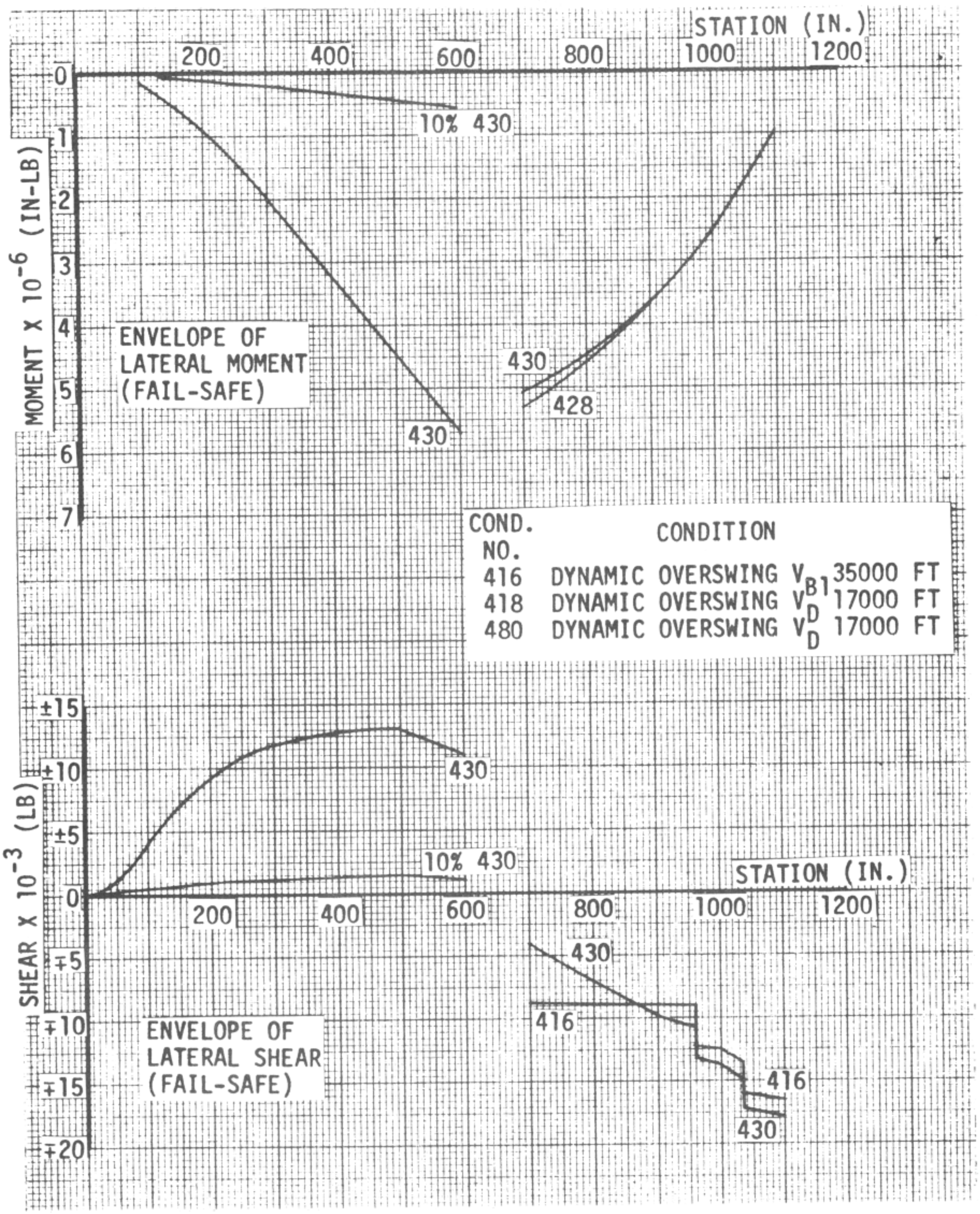


FIGURE 4-14. ENVELOPE OF LATERAL FAIL-SAFE SHEAR AND MOMENT (DC-9-31)

STRESS CALCULATIONS FOR DAMAGED STRUCTURE

The general equation for bending stresses is:

$$f_b = \frac{M_y I_{yz} - M_z I_y}{I_y I_z - I_{yz}^2}(Y) + \frac{M_z I_{yz} - M_y I_z}{I_y I_z - I_{yz}^2}(Z) \qquad \text{Ref. 21}$$

where

f_b = stress at point (y,z) (psi)

M_y = moment about y axis (in-lb)

M_z = moment about z axis (in-lb)

I_y = moment of inertia about y axis ($in.^4$)

I_z = moment of inertia about z axis ($in.^4$)

I_{yz} = product of inertia ($in.^4$)

Y = distance along y axis (in.)

Z = distance along z axis (in.)

The section properties for the basic shell are shown in Table 4-7. The section properties for six variations of damaged structure are shown in Table 4-8. Bending stresses are calculated for these configurations using the bending moments from Table 4-6 and the section properties from Table 4-8.

TABLE 4-7. DC-9-31 FUSELAGE GEOMETRY AND SECTION PROPERTIES FROM STATION 300 TO 400

LONGERON	Y in.	Z in.	ds in.	LONG AREA in.^{2}	SKIN AREA TEN in.^{2}	SKIN AREA COMP in.^{2}	TOTAL AREA TEN in.^{2}	TOTAL AREA COMP in.^{2}
1	0	92	7.55	.127	.239	.069	.366	.196
2	7.5561	91.5648		.112	.378	.075	.490	.187
3	15.0123	90.2649		.112	.378	.075	.490	.187
4	22.2699	88.1176		.112	.378	.075	.490	.187
5	29.2330	85.1511		.112	.378	.075	.490	.187
6	35.8094	81.4049		.112	.378	.075	.490	.187
7	41.9123	76.9283		.112	.378	.075	.490	.187
8	47.4608	71.7807		.112	.378	.075	.490	.187
9	52.3816	66.0300		.141	.378	.075	.519	.216
10	56.6096	59.7524	7.55	.141	.378	.075	.519	.216
11	60.0889	53.0309	7.00	.249	.614	.359	.863	.608
12	61.52	46.22		.203	.442	.358	.645	.561
13	62.95	39.41		0	0	0	0	0
14	64.38	32.62		.203	.442	.358	.645	.561
15	65.8109	25.8106	8.30	.249	.683	.404	.932	.653
16	65.2407	17.5350	8.30	.112	.523	.119	.635	.231
17	63.6343	9.3982		.112	.576	.119	.688	.231
18	60.3772	0	7.10	.558	.492	.245	1.050	.803
19	59.6590	-6.8686		.227	.355	.075	.582	.302
20	56.1622	-13.4941		.141	.355	.075	.496	.216
21	52.9199	-19.7883		.141	.355	.075	.496	.216
22	48.9751	-25.6679		.141	.355	.075	.496	.276
23	44.3803	-31.0547		.141	.355	.075	.496	.276
24	39.1963	-35.8772		.141	.355	.075	.496	.276
25	33.4921	-40.0715		.141	.355	.075	.496	.276
26	27.3433	-43.5818		.324	.411	.131	.735	.455
27	20.8315	-46.3616		.141	.355	.075	.496	.216
28	14.0432	-48.3739		.141	.355	.075	.496	.216
29	7.0685	-49.5921	7.10	.198	.355	.075	.553	.273
30	0	-50.		.099	.178	.038	.277	.137

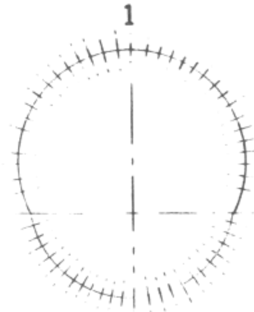


TABLE 4-8. SECTION PROPERTIES OF PARTIAL SHELL STATION 300 TO 400

	SECTION DESCRIPTION	Z in.	Y in.	ΣA in.^{2}	I_{yz} in.^{4}	I_{zz} in.^{4}	I_{yy} in.^{4}
	FULL	31.66	0	26.36	0	63901.64	55462.17
	LONG L.H. 6-9 REMOVED	28.22	-3.63	24.37	-3974.85	59563.04	51554.55
	LONG L.H. 5-11 REMOVED	23.90	-8.86	22.01	-8499.44	52715.74	46749.90
	LONG L.H. 1-14 REMOVED	16.77	-15.11	19.01	-12533.50	44855.00	37616.13
	LONG L.H. 1-14 & R.H. 2-5 REMOVED	8.49	-14.72	17.05	-12034.92	44701.39	26273.38
	LONG L.H. 1-14 & R.H. 2-9 REMOVED	-.16	-10.78	15.06	-7716.99	42626.13	16566.92
	LONG L.H. 1-14 & R.H. 2-14 REMOVED	-10.50	0	12.39	0	34487.20	8858.53

EXAMPLE: Bending stresses at longeron 15 with longerons 1 through 14 on both sides missing

$$f_b = \frac{(-5234015)(0) - (0)(8858.53)}{(8858.53)(34487.20) - 0} (65.8109)$$

$$+ \frac{(0)(0) - (5234015)(34487.20)}{(8858.53)(34487.20) - 0} (25.81 + 10.50)$$

$$= 21454 \text{ psi}$$

The results are plotted Figure 4-15.

The general equation for the change in shear flow caused by direct shear is:

$$\Delta q = \left[\frac{V_z I_{yz} - V_y I_y}{I_y I_z - I_{yz}^2} (Y) + \frac{V_y I_{yz} - V_z I_z}{I_y I_z - I_{yz}} (Z) \right] A\ell \quad \text{Ref. 21}$$

qhere

Δq = change in shear flow (lb/in)

A_ℓ = longeron area (in.^2)

Other terms as before.

The change in shear is calculated for the skin panel between longeron 15 and 16 for the vertical and lateral load cases in Table 4-6, using the section properties in Table 4-8. (Again assuming longerons 1 through 14 on both sides are missing.) Since longeron 15 is at the edge of the remaining section, the shear flow is equal to the change in shear flow ($q = \Delta q$).

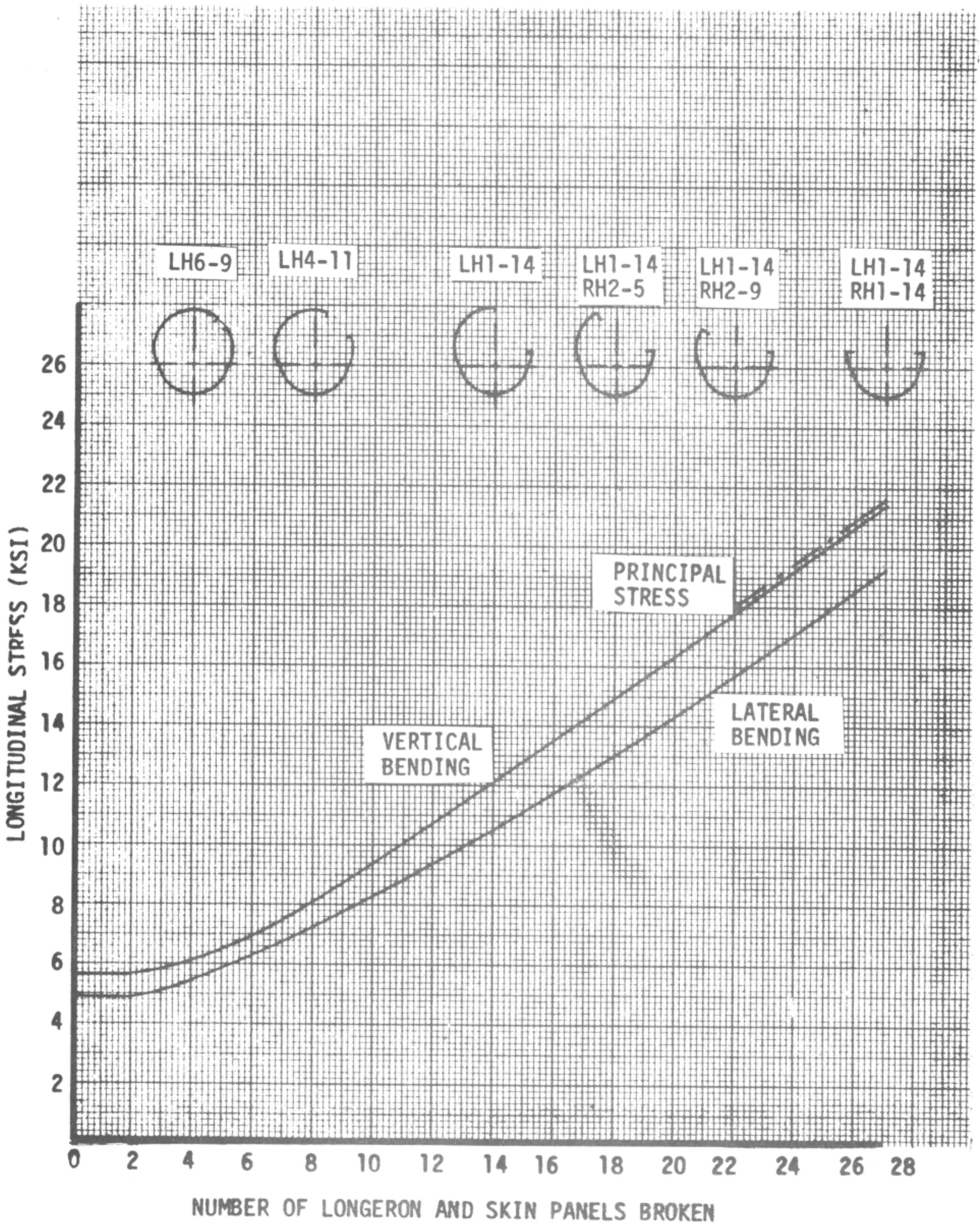


FIGURE 4-15. LONGITUDINAL STRESS AT UPPER EDGE OF HOLE AT STATION 400

VERTICAL CASE:

$$q = \frac{(-23422)(0) - (0)(8858.53)}{(8858.53)(34487.20) - 0} (65.8109)$$

$$+ \frac{(0)(0) - (-23422)(34487.20)}{(8858.53)(34487.20) - 0} (25.81 + 10.50) \quad (.932) = 89 \text{ lb/in}$$

The corresponding shear stress in .063 skin =

$$f_s = \frac{q}{t} = \frac{89}{.063} = 1420 \text{ psi}$$

LATERAL CASE:

$$q = \frac{(-20367)(0) - (+1272)(8858.53)}{(8858.53)(34487.20) - 0} (\pm 65.8109)$$

$$+ \frac{(+1272)(0) - (-20367)(34487.20)}{(8858.53)(34487.20) - 0} (25.81 + 10.50) \quad (.932)$$

$$= 80 \text{ lb/in}$$

The corresponding shear stress in .063 skin

$$f_s = \frac{80}{.063} = 1271 \text{ psi}$$

The fuselage is normally assumed to carry the torsional moment as a single cell neglecting the floor's contribution. With a hole in the upper skin this section would no longer be a closed cell. Open cells have poor torsional qualities. The torque would have to be carried by differential bending. Instead for this analysis, the closed cell bounded by the floor and lower skin is assumed to carry all of the torsion and differential bending is neglected.

The lateral load condition produces the highest shear flow. The torque consists of the torsional moment from Table 4-6 plus the torque caused by the side shear not being at the shear center of the section (Figure 4-16).

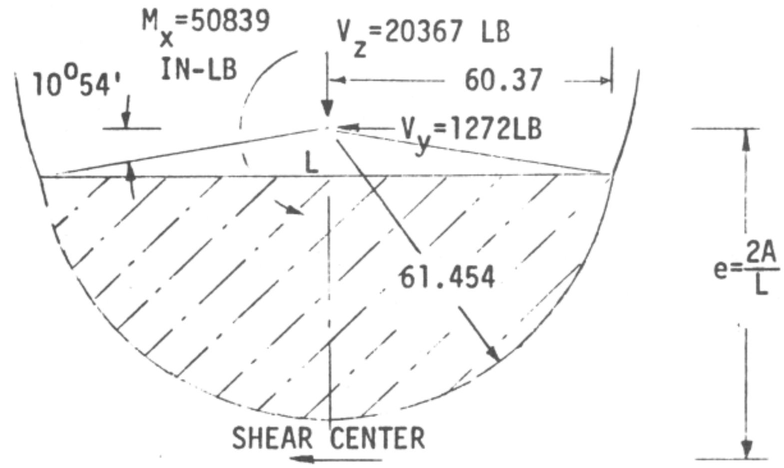


FIGURE 4-16. TORQUE CARRYING STRUCTURE

Side shear is located approximately at point "O" as shown in Figure 4-16.

$$\text{ENCLOSED AREA} = A = \pi\ (61.454)^2\ \frac{180°-(2)(10°54')}{360} - (11.454)(61.454)$$
$$= 4510.\ \text{in}^2$$

$$\text{ECCENTRIC} = e = \frac{2A}{L} = \frac{2(4510)}{2(60.37)} = 74.7\ \text{in.}$$

$$\text{TORQUE} = T = M_x+V_ye = 50839+(1272)(74.7) = 145862.9\ \text{in-lb}$$

$$\text{SHEAR FLOW} = q = \frac{T}{2A} = \frac{1458629}{2(4510)} = 16\ \text{lb/in}$$

This is a very low shear flow and is satisfactory by inspection.

The critical stress occur in the last remaining skin panel adjacent to the damaged area. Per the above calculations, the maximum stresses in the longeron 15 to 16 skin panel (all the longerons above missing) are f_b = 21454 psi and f_s = 1420 psi. These are combined to obtain the maximum normal (principal) stress according to the equation:

$$f_{max} = f_b/2 + \sqrt{(f_b/2)^2 + f_s^2} \qquad \text{Ref. 22}$$

$$= 21454/2 + \sqrt{(21454/2)^2 + 1420^2} = 21548 \text{ psi}$$

The principal stresses are added to Figure 4-15 to provide the limiting situation. Principal stresses are compared to the residual strength of the structure to determine the permissible damage.

RESIDUAL STRENGTH

The ultimate strength of the fuselage section is 64000 psi tension for the upper longeron-skin panel and 35000 psi compression for the lower longeron-skin panels.[1] However, when the fuselage is damaged by an explosion many torn and jagged edges exist. The ultimate is greatly reduced and is governed by the fracture toughness of the material.

For centrally cracked wide panels, fracture toughness is

$$K_c = \sigma_R \sqrt{\pi a}$$

where

K_c = plane stress fracture toughness psi $\sqrt{in.}$
σ_R = residual gross area stress psi
a = critical half crack length in.

The residual strength of an unstiffened panel can be determined thus

$$\sigma_R = \frac{K_c}{\sqrt{\pi a}}$$

If σ_R were plotted against half crack length a, the curve would be as shown in Figure 4-17.

The effect of stiffening can be included by introducing a term R_{ct} where

$$R_{ct} = \frac{\text{crack tip stress in unstiffened panel}}{\text{crack tip stress in stiffened panel}}$$

Thus for a stiffened panel the equation would become

$$\sigma_R = \frac{K_c R_{ct}}{\sqrt{\pi a}}$$

Poe[23] has determined R_{ct} as a function of rivet pitch, longeron spacing and % of stiffening area. R_{ct} increases as the crack tip approaches a frame with transfer of load from the cracked sheet into the frame. If σ_R were plotted against crack length, the curve would be as shown in Figure 4-18.

The significance of this curve is a follows: If a gross stress of σ_R is applied with a half crack length a_A, fast fracture will occur at A and the crack will be arrested at B. The residual strength of the panel is σ_c and any fast fracture at stress levels higher than this would not be arrested.

The above equations are for centrally cracked structure. A correction factor may be included to account for the free surface when the structure is subjected to an edge crack

$h(a/b)$ = correction factor for edge crack as a function of half crack length to half panel width

= 1.12 for small cracks in wide panels[24]

Then the residual stress equation becomes:

$$\sigma_R = \frac{K_c\ R_{ct}}{h(a/b)\ \sqrt{\pi a}}$$

This equation gives the residual strength for stiffened structure with an edge crack. The half crack length refers to skin damage beyond the last intact longeron as shown in Figure 4-19. The resulting residual strength calculations for the top and side panels are shown in Table 4-19.

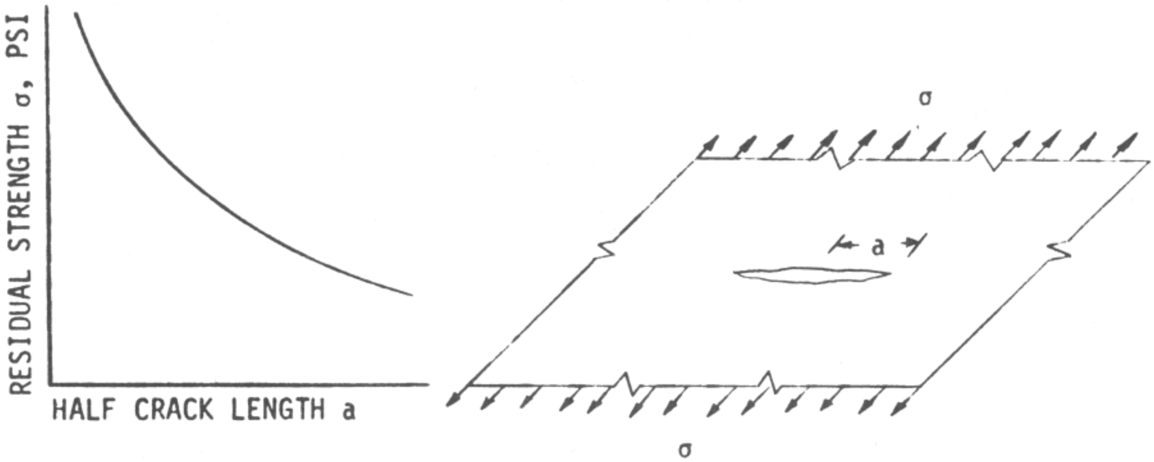


FIGURE 4-17. RESIDUAL STRENGTH FOR UNSTIFFENED PANEL

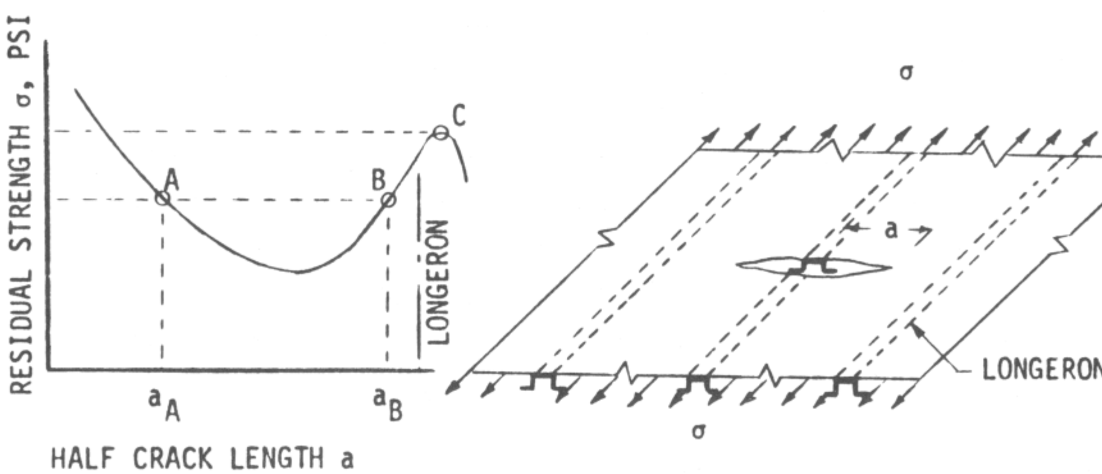


FIGURE 4-18. RESIDUAL STRENGTH FOR STIFFENED PANEL

TABLE 4-9. RESIDUAL STRENGTH CALCULATIONS FOR TOP AND SIDE PANELS

PANEL	PANEL THICKNESS (IN.)	HALF CRACK LENGTH (IN.)	% STIFFENING $\frac{\text{LONG AREA}}{\text{TOTAL AREA}}(100)$ (REF TABLE 4-7)	R_{ct} (REF 23)	$\sigma_R = \frac{K_c R_{ct}^*}{h(a/b)\sqrt{\pi a}}$ PSI
TOP	.050	8.30	23	2.00	23,633
SIDE	.063	9.13	18	1.79	20,173
SIDE	.063	7.47	18	1.33	16,570
SIDE	.063	6.64	18	1.27	16.783
SIDE	.063	5.81	18	1.25	17,660
SIDE	.063	4.15	18	1.24	20,727
SIDE	.063	3.73	18	1.24	21,849
SIDE	.063	3.32	18	1.24	23,174

* K_c = 67,600 psi$\sqrt{\text{IN}}$ REF 25

$h(a/b)$ = 1.12 REF 24

The top panel extends down to longeron 11. The residual strength for this panel is high (23633 psi, Table 4-19) relative to the longitudinal stress when the damage is limited to the top panel as seen in Figure 4-15. The window belt is between longeron 11 and 15. It consists of window cut-outs, skin and heavy doublers. Remaining intact members can be loaded to the ultimate strength of the material. The skin is spliced at longeron 15. If the longeron 15 and the skin panels below are undamaged, the allowable stress is 64,000 psi[8] while the longitudinal stress is 21,548 psi. It is only when the damage gets below longeron 15 that the allowable stress is less than the longitudinal stress. The side panel allowable strength from Table 4-19 is plotted in Figure 4-20 along with the longitudinal stress in that panel (the structure above longeron 15 is assumed destroyed).

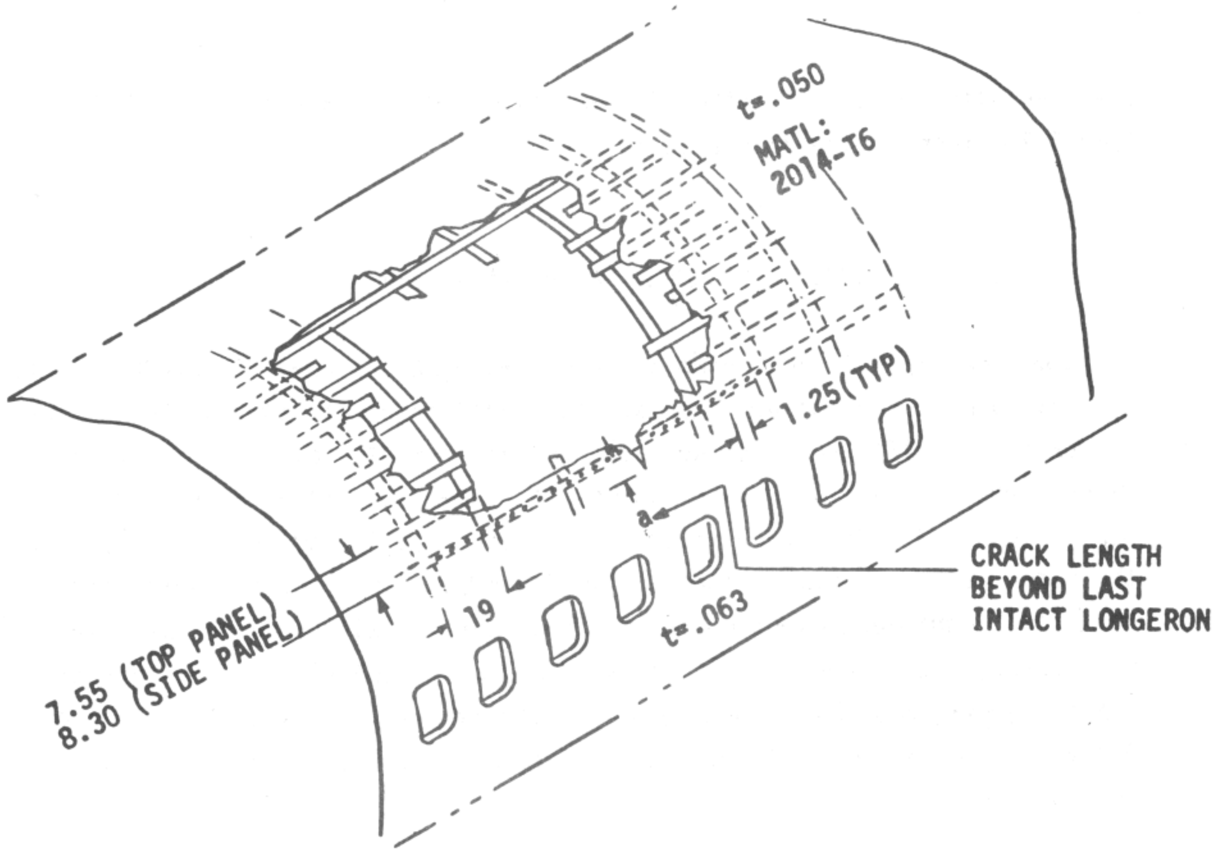


FIGURE 4-19. ILLUSTRATION OF BOMB DAMAGE

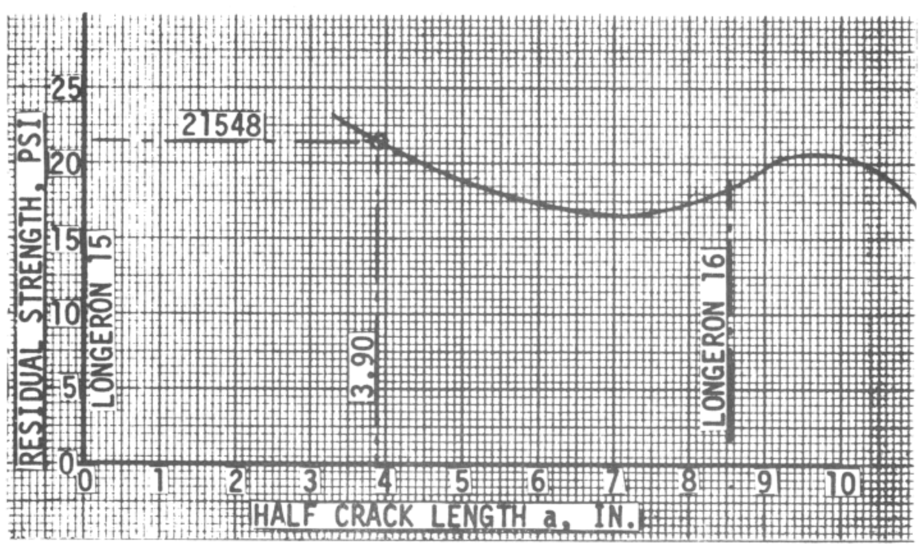


FIGURE 4-20. RESIDUAL STRENGTH FOR SIDE PANEL

The capability of the structure is exceeded when the skin is cracked more than 3.90 inches below longeron 15.

SUMMARY OF DAMAGE AND RESIDUAL STRENGTH

In summation, a 4-lb. 40% dynamite bomb detonated in the left hand hat rack will damage the top of the fuselage from left hand longeron 11 to right hand longeron 7. Sufficient residual strength will exist even if all the structure above longeron 15 is removed when it is exploded forward of station 400. See Figure 4-21. The floor appears to be outside the damage area. However, the floor is the crucial element. The floor acts as a barrier protecting the vital equipment from the blast. The floor is vulnerable to this kind of loading. Every effort must be made to protect the floor. The seat backs should be reclined and loose equipment and baggage should be piled on the floor in the region of the direct blast to protect the floor from the initial shock waves.

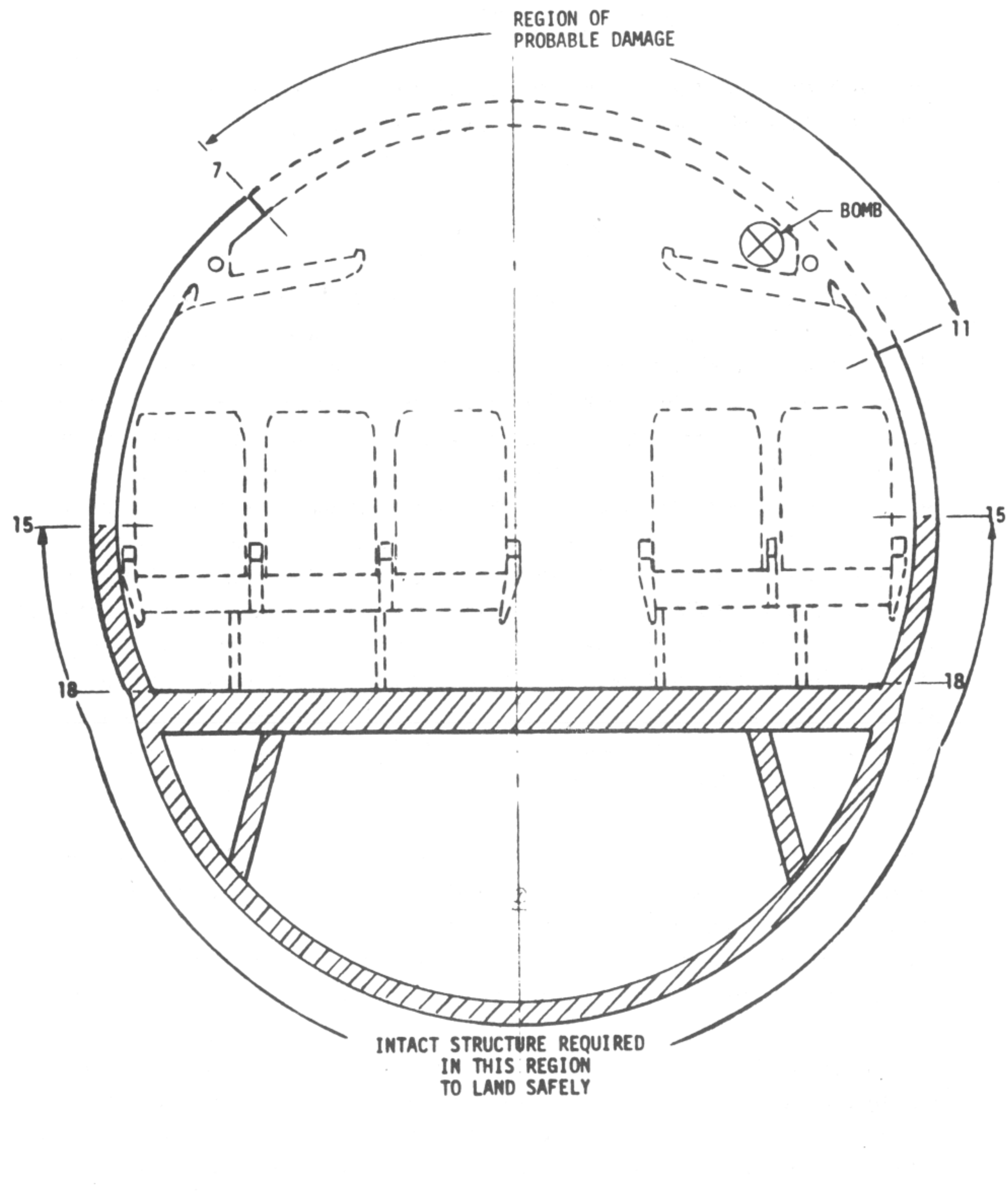


FIGURE 4-21. ESTIMATED DAMAGE AND REQUIRED STRUCTURE FOR 4-LB 40-PERCENT DYNAMITE EXPLOSION IN HAT RACK FORWARD OF STATION 400

5 BOMB IN THE TAIL SECTION

This section calculates the damage done to the structure if 4-lb. 40% dynamite bomb is detonated in the tail cone or on the ventral stairs.

BOMB IN THE TAIL CONE

The tail cone is a logical safe bomb location because much of the local structure only acts as a fairing and, as such, just supports aerodynamic loads. The upper portion of the rear spar and rudder control support bulkheads are directly in front of the blast. Figure 5-1 shows these bulkheads and the critical cables. If the blast destroys these bulkhead webs, the control systems can be assumed inoperable. The cables represent very small targets for the blast and would only be damaged by bomb container fragmentation. The same is true of the hydraulic lines and electrical wiring. The airplane can still land in an emergency with loss of the elevator and rudder control as long as the horizontal stabilizer trim and ailerons are still useable. The rear spar bulkhead is a primary load path for the tail loads. The upper portion is used to shear the tail loads into the fuselage. However, there is a redundant load path at the aft intermediate spar bulkhead. The airplane could probably land even without the rear spar bulkhead. The bomb should be placed as far aft in the tail cone as possible.

Figure 5-2 shows the location of the bomb in the tail cone relative to the structure. Points at one foot radial distance are noted. The critical impulse and ¼ natural period of the structure are compared to the reflective impulse and blast time at each of the 14 locations. The method is outlined in Section 4. The results are shown and the damage is noted in Table 5-1.

The damage is limited to the structure behind the rear spar bulkhead. The rudder surface is oblique to the blast and only the lower corner will be damaged. The cone and its attachment ring will be destroyed. The upper web of the rudder control support bulkhead will be damaged and the rudder will probably be inoperative. The top and side skin panels aft of the rear spar bulkhead will be damaged. The upper rear spar bulkhead is just out of the

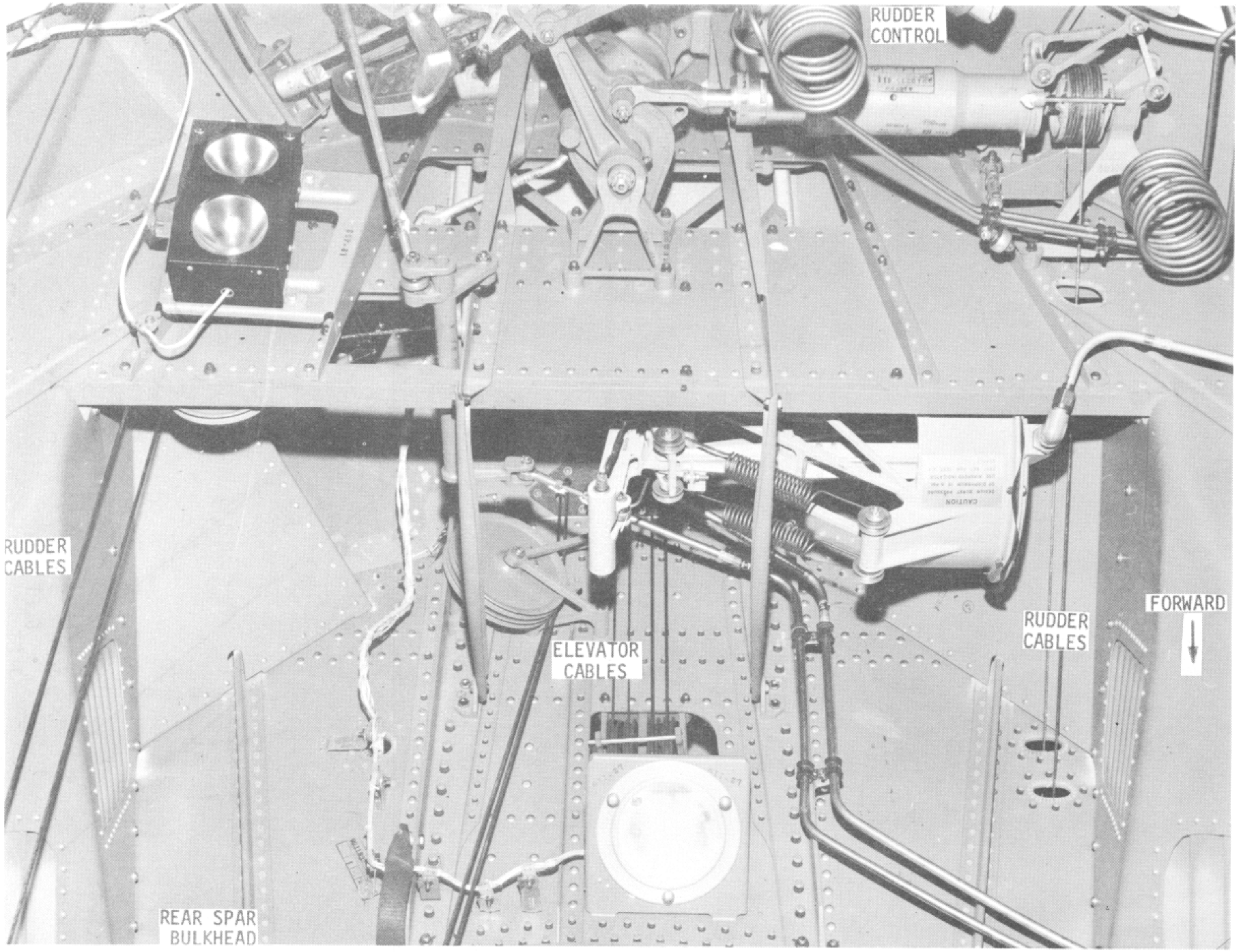


FIGURE 5-1. VIEW FROM TAIL CONE ACCESS DOOR LOOKING UP AND FORWARD

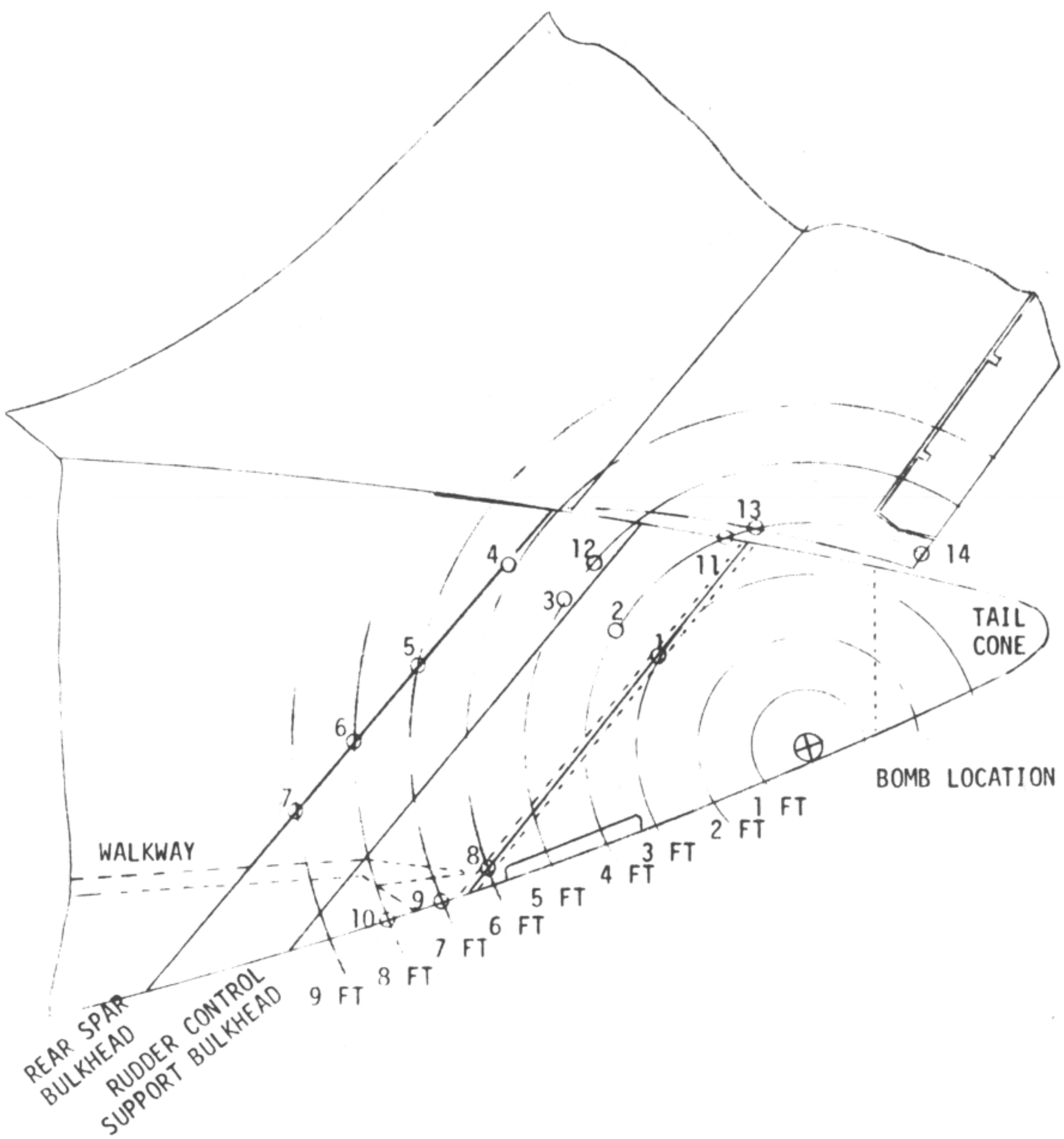


FIGURE 5-2. TAIL CONE BOMB LOCATION

TABLE 5-1. DAMAGE CALCULATIONS FOR 4-LB 40-PERCENT DYNAMITE BLAST LOCATED IN TAIL SECTION

LOCATION (FIG 5-1& -2)	THICKNESS IN.	PANEL SIZE IN.	RADIAL DISTANCE FT.	CRITICAL IMPULSE $I_c=\rho t V_c$ psi-ms	¼ NATURAL PERIOD $T_N/4$ ms	ANGLE OF INCIDENCE α DEG	REFLECTIVE IMPULSE $I_{r-\alpha}$ psi-ms	DURATION OF POS. PULSE t_o ms (FIG 4-5)	REMARKS
TAIL CONE									
1 SIDE PANEL	.032	11xLONG	4	23.9	4.4	68	30	.60	POSSIBLE DAMAGE
2 SIDE PANEL	.032	11xLONG	5	23.9	4.4	70	27	.96	POSSIBLE DAMAGE
3 SIDE PANEL	.032	11xLONG	6	23.9	4.41	71	26	1.3	POSSIBLE DAMAGE
4 REAR SPAR BKHD	.080	6x15	6	59.7	.49	0	18	1.3	$I_{r-\alpha}<I_c, t_o>T_N/4$
5 REAR SPAR BKHD	.050	9x12	7.6	37.3	1.41	30	41	1.7	$t_o>T_N/4$
6 REAR SPAR BKHD	.032	7x10	8.5	23.9	1.41	41	41	1.9	$t_o>T_N/4$
7 REAR SPAR BKHD	.032	4x11	9.5	23.9	.56	48	35	2.1	$t_o>T_N/4$
8 LOWER PANEL	.032	12x18	6	23.9	4.35	90	18	1.3	$I_{r-\alpha}<I_c$
9 LOWER PANEL	.032	12xLONG	7	23.9	5.25	90	16	1.6	$I_{r-\alpha}<I_c$
10 LOWER PANEL	.032	12xLONG	8	23.9	5.25	90	14	1.8	$I_{r-\alpha}<I_c$
11 TOP PANEL	.063	11x18	4	47.0	1.91	30	100	.60	POSSIBLE DAMAGE
12 RUDDER CONTR BKHD	.040	9x10	5	29.9	1.55	0	74	.96	POSSIBLE DAMAGE
13 RUDDER RIB	.032	5xLONG	4	23.9	.91	38	100	.60	$t_o>T_N/4$
14 RUDDER PANEL	.040	8x8	4	29.9	1.16	90	21	.60	POSSIBLE DAMAGE
VENTRAL STAIRS									
1 JAMB WEB	.040	3xLONG	2	29.9	.26	42	78	.075	POSSIBLE DAMAGE
2 LOWER PANEL	.032	12xLONG	2	23.9	5.25	69	34	.125	POSSIBLE DAMAGE
3 LOWER PANEL	.032	12xLONG	3	23.9	5.25	85	25	.25	POSSIBLE DAMAGE
4 SIDE PANEL	.032	11xLONG	4	23.9	4.41	90	21	.60	$I_{r-\alpha}<I_c, t_o>T_N/4$

See Page 85 for Damage Criteria.

blast effective zone and will remain intact. The elevator control cables might be damaged. They are redundant and elevator control may be maintained.

If the bomb is detonated in the tail cone, the fiberglass cone will be blasted away and most of the gas products will be vented overboard. Only 11 percent of the blast sphere would be available to pressurize the approximately 2000 Ft^3 tail section volume. The charge-colume ratio is very low and no significant long duration pressure is anticipated.

The best place to put a bomb to do the least damage is outside the skin for three reasons: 1) The structure near the blast acts as protection for the opposite side. 2) Curved surfaces are convex to the blast so that the angle of incidence decreases sinusoidally as the angular distance increases from the projected normal to the blast. The reflective impulse received by the structure is a step function at about 40 degrees. At angles larger than 40 degrees, the reflective impulse is greatly reduced. The opposite extreme exists when the blast emanates from the center of the circle. Then every point on the surface is normal to the blast. 3) The gas products do not charge the enclosure with a long duration, static, pressure.

The bomb could be placed on the stairs and extended into the slipstream. This represents an ideal solution. The bomb would be outside the shell in the region of non-critical structure. The bomb would be more than nine feet from the trailing edge of the engines, far from the fuel lines, electrical wires, plumbing, people and most control cables. The tail section walkway is the ceiling of the stairway tunnel. It could be lowered to absorb impulse and reduce fragmentation. The stairs are expendable.

Two important components are in the vicinity. The rudder control cables and one redundant set of elevator cables are routed along the lower skin. Their pulley bracket installation next to the stairs is shown in Figure 5-3. But again, the airplane can land without these controls.

The lower part of the vertical stabilizer rear spar bulkhead terminates at the stairwell jamb installation. Only the upper part of this bulkhead is required to shear the tail loads into the skin. Even there, redundance is provided in the form of an intermediate spar bulkhead.

The bomb should be placed on the aft most step in order to get it as far away from the shell as possible. A crew member should stand on the stairs as the stairs are powered down in order to lift the folding step. The bomb should be placed on the step made accessible by lifting the folding step. Figure 5-4 shows the location of the bomb on the stairs relative to the structure.

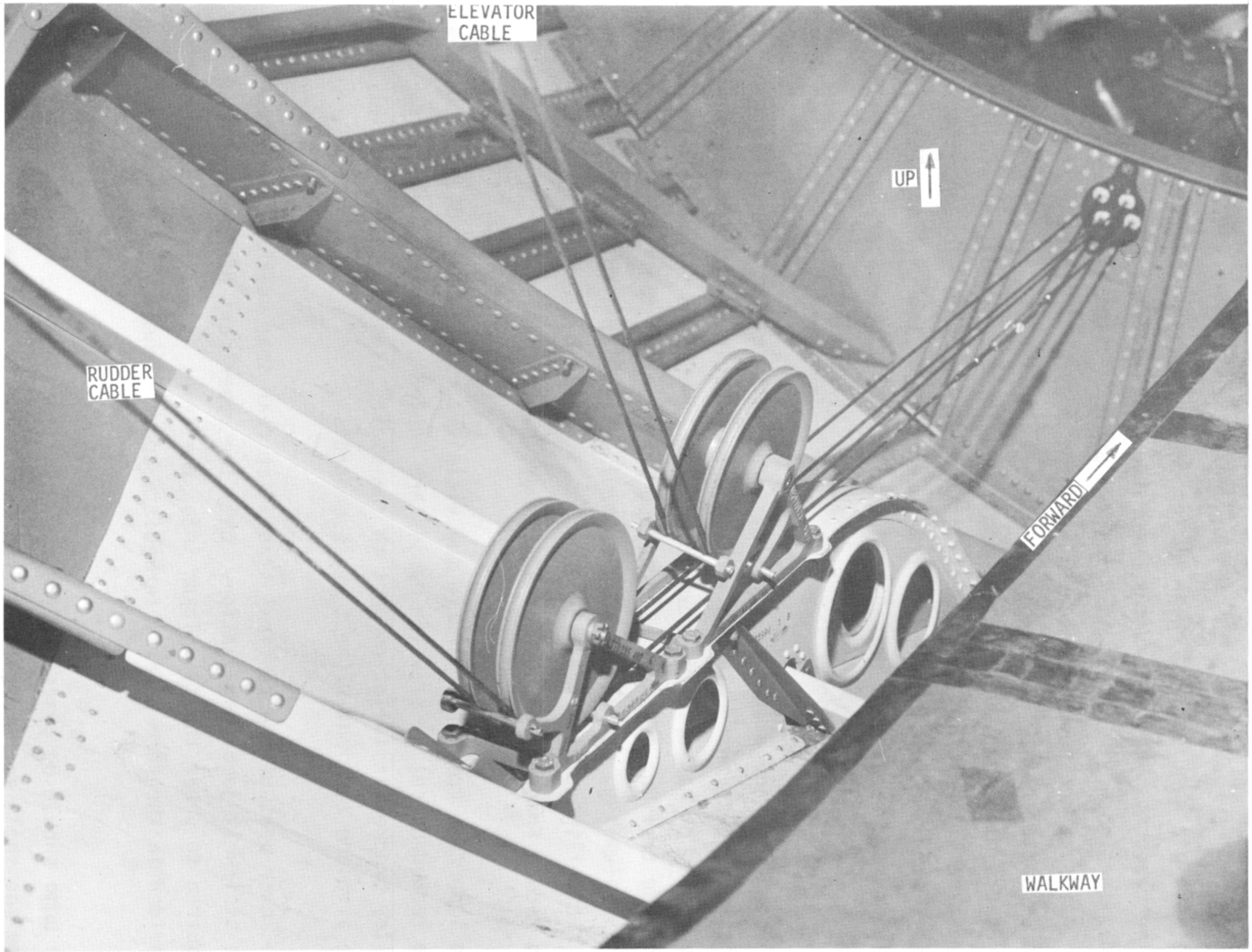


FIGURE 5-3. VIEW OF CONTROL PULLEYS NEXT TO VENTRAL STAIRS

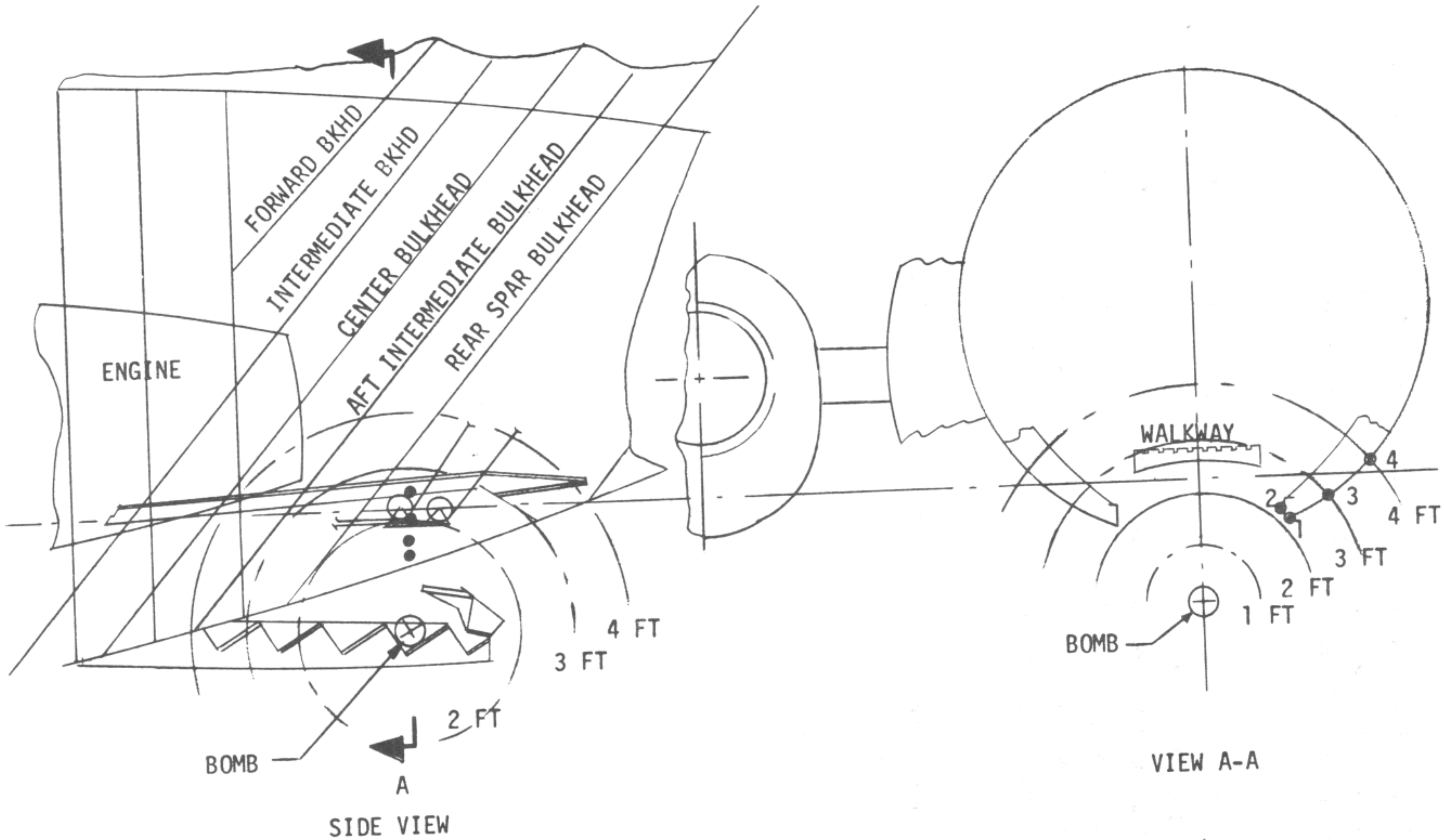


FIGURE 5-4. VENTRAL STAIRS BOMB LOCATION

Points at one foot radial distances are noted. The critical impulse and ¼ natural period of the structure are compared to the reflective impulse and blast pulse time at four locations. The results are shown and the damage is noted in Table 5-1.

View A-A in Figure 5-4 illustrates the advantage of putting the bomb on the stairs outside the shell. No shell structure is within two feet and much of the structure four or more feet from the bomb is "shaded" and protected from the direct blast. Damage will be slight. The stairs, walkway, side jambs, fiberglass tunnel walls and local skin will probably be destroyed. The control pulley bracket will probably be damaged and the control cables inoperative.

The estimated damage done for both the tail cone and stairs explosions are sketched in Figure 5-5.

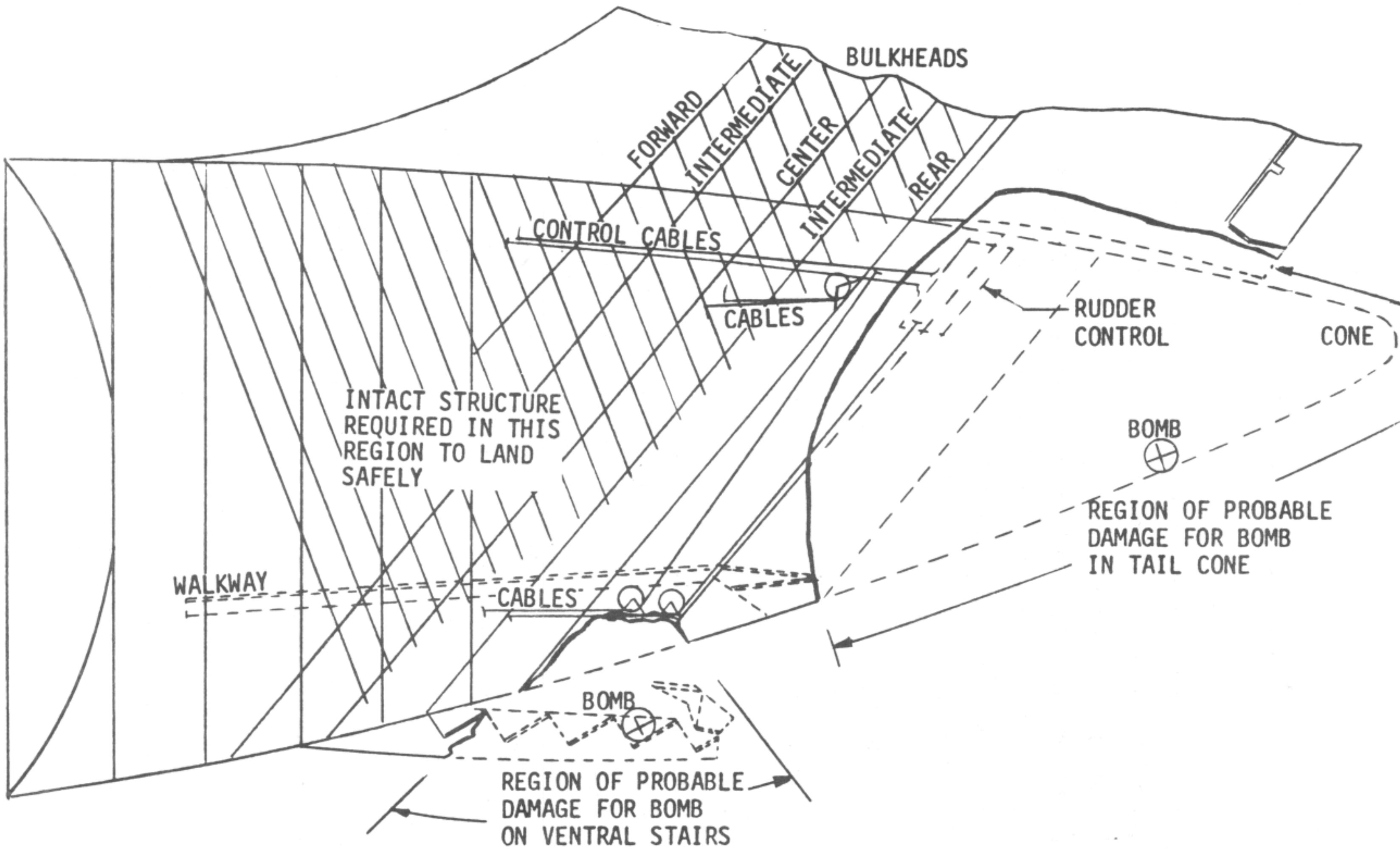


FIGURE 5-5. ESTIMATED DAMAGE AND REQUIRED STRUCTURE FOR 4-LB 40-PERCENT DYNAMITE EXPLOSION IN TAIL SECTION

6 PREVIOUS DAMAGE EXPERIENCE

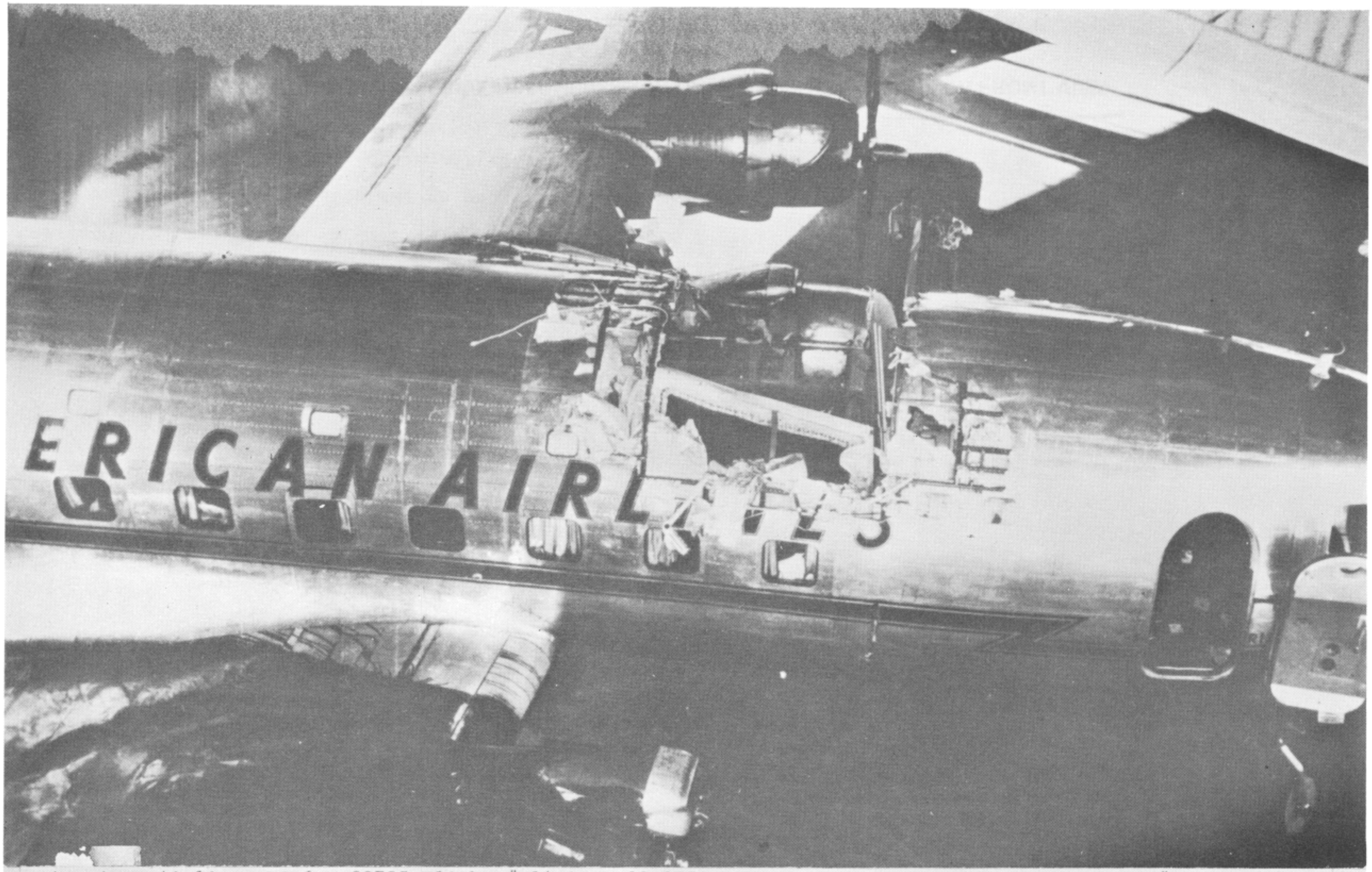
There have been many incidents where bombs, broken engine parts or propellers have damaged the fuselage and caused explosive decompression. Some of these incidents have caused catastrophic failure resulting in loss of the aircraft and in others, sufficient residual strength has remained such that the aircraft was able to land safely. This section reviews three examples of massive structural damage where sufficient residual strength remained to land the aircraft safely.

AMERICAN AIRLINES N90705 DC-6

A propeller blade from the number 3 engine failed on American Airlines N90705 DC-6 flight #14 on August 22, 1950. (Douglas Serial #42858).

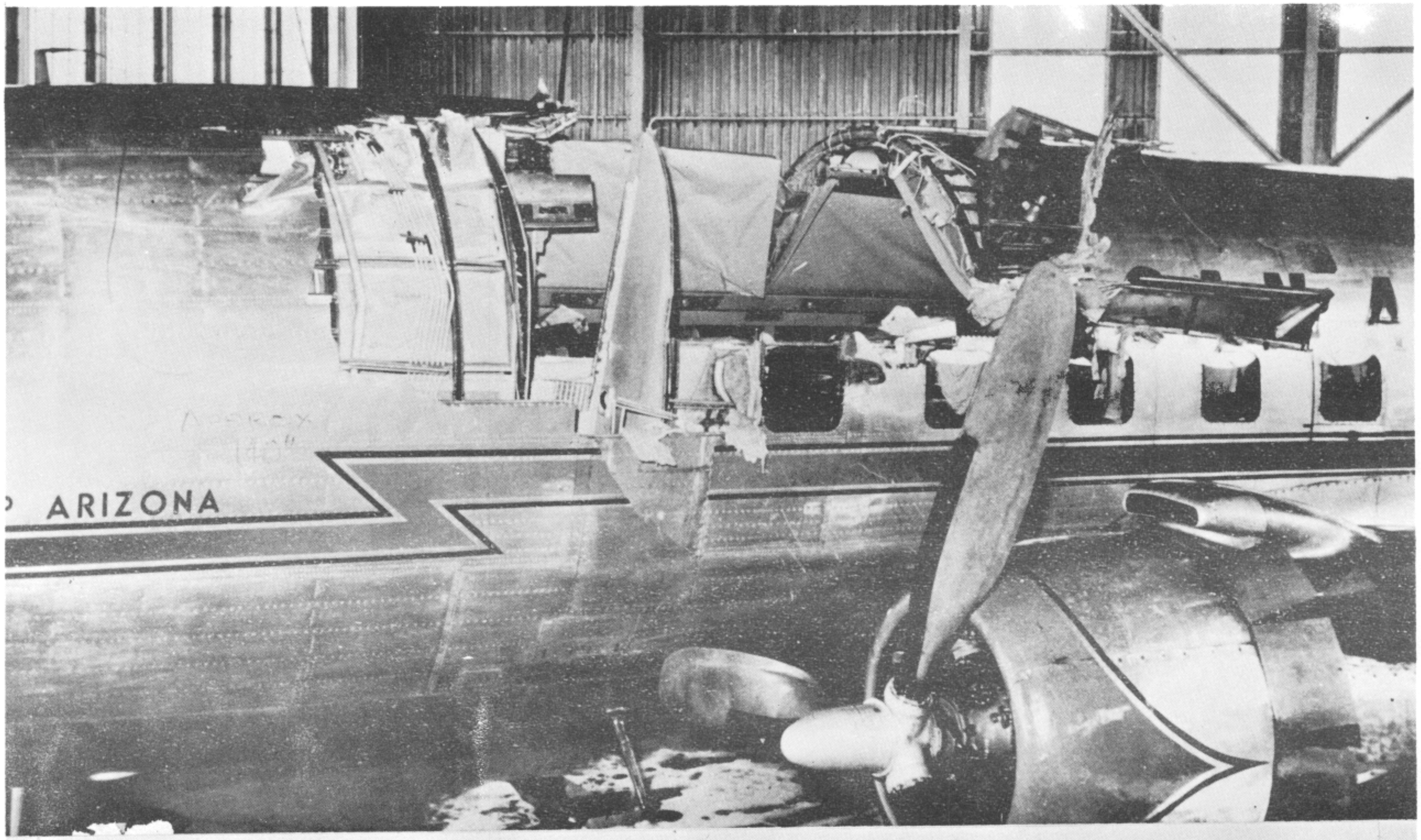
The incident occurred near Eagle, Colorado on a flight from Los Angeles, California to Chicago, Illinois. A portion of the propeller blade pierced the fuselage cabin resulting in explosive decompression. Almost immediately the #3 engine wrenched free and fell from the aircraft. A safe emergency landing was made 19 minutes later at Denver, Colorado. Five passengers and one stewardess sustained minor injuries and one passenger died from heart failure out of a total of 55 passengers and a crew of 5.

At the time of failure the aircraft was cruising at an indicated altitude of 21,000 feet with approximate cabin altitude of 8600 feet resulting in a pressure differential of 4.16 psi. The blade portion struck the fuselage edgewise and left the other side flatwise. The practically instantaneous depressurization caused other cabin damage such as loosening the sound proof lining. The boundaries of the opening in the fuselage, about 250 square feet were jagged and torn. Pictures of the aircraft are shown in Figures 6-1, 6-2 and 6-3. It can be seen that about 50% of the bending section of the aircraft was removed by the explosion.



American Airlines DC-6 N 90705 Flight # 14 Aug. 22 1950 Photo # SM 1678 71
After explosive decompression due to propeller blade piercing the fuselage.
Aircraft altitude 21000 ft., cabin altitude 8600 ft., differential pressure 4.16 psi

FIGURE 6-1. FLIGHT NO. 14 EXPLOSIVE DECOMPRESSION DAMAGE – TOP VIEW

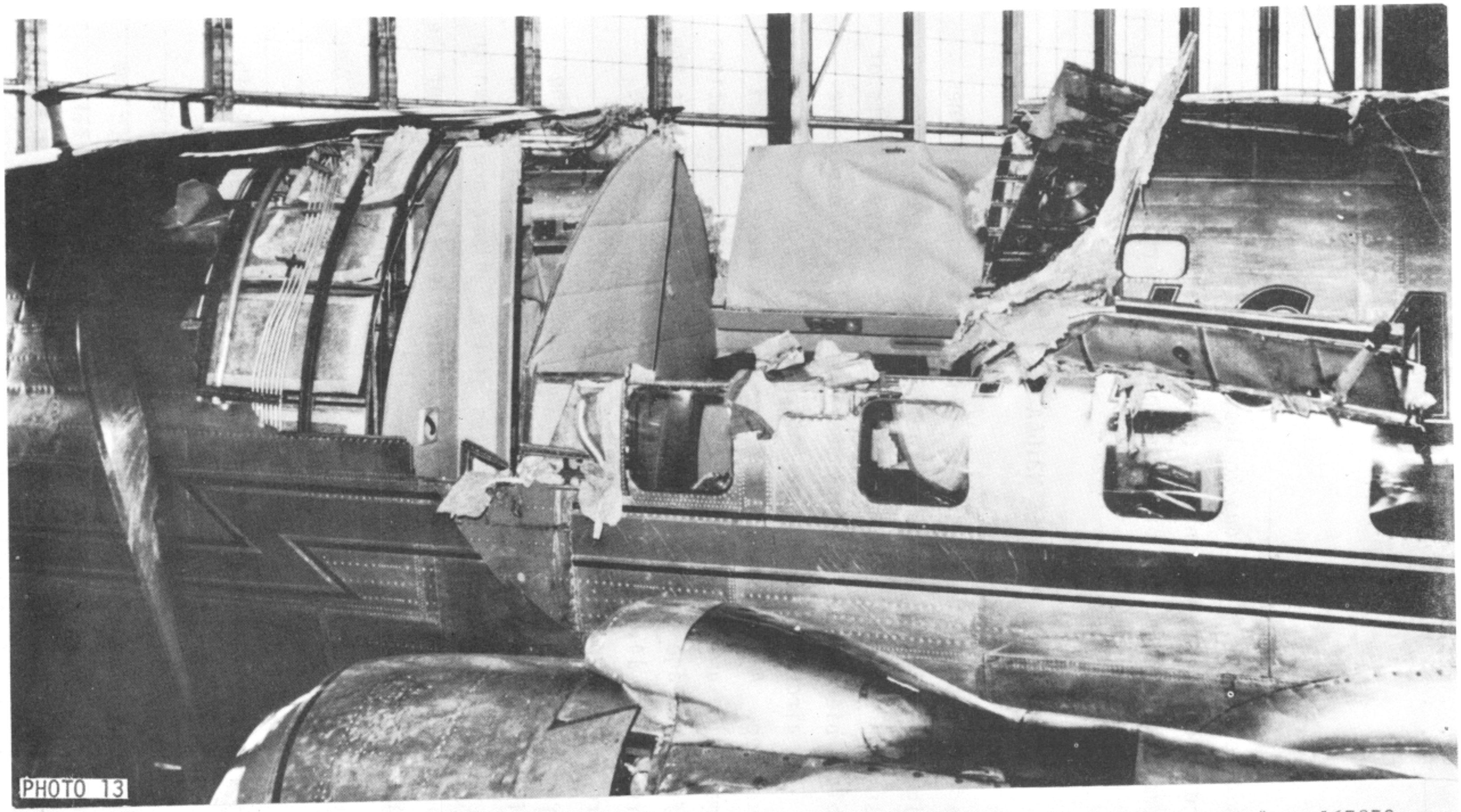


American Airlines DC-6 N 90705 Flight # 14 Aug. 22 1950 Photo # SM 167873

After explosive decompression due to propeller blade piercing the fuselage.

Aircraft altitude 21000 ft., cabin altitude 8600 ft., differential pressure 4.16 psi.

FIGURE 6-2. FLIGHT NO. 14 EXPLOSIVE DECOMPRESSION DAMAGE – THREE-QUARTER FRONT VIEW



American Airlines DC-6 N 90705 Flight # 14 Aug. 22 1950 Photo # SM 167872

After explosive decompression due to propeller blade piercing the fuselage.

Aircraft altitude 21000 ft., cabin altitude 8600 ft., differential pressure 4.16 psi.

FIGURE 6-3. FLIGHT NO. 14 EXPLOSIVE DECOMPRESSION DAMAGE — SIDE VIEW

The landing was made without flaps due to loss of actuator hydraulic fluid. Reversed pitch on the remaining 3 propellers and an application of emergency compressed air to the brakes stopped the aircraft on the runway.

AMERICAN AIRLINES N316AA DC-7

A propeller blade from the number 1 engine failed on American Airlines N316AA DC-7 flight 87 on March 5, 1957. The incident occurred near Memphis, Tennessee on a flight from New York to San Francisco. Parts of the propeller blade pierced the fuselage cabin resulting in explosive decompression. The aircraft made an uneventful landing 7 minutes later at Memphis, Tennessee. All passengers and crew members were affected by sudden decompression and 5 passengers received contusions, lacerations and abrasions from fuselage parts and debris. At the time of failure the aircraft was cruising at 14,000 feet with cabin differential pressure 5.1 psi. The resulting explosive decompression tore a jagged and irregular shaped hole approximately 17 feet long by 4 feet wide in the upper forward fuselage. There was considerable internal damage in the cabin as a result of decompression. Pictures of the failure are shown in Figures 6-4 through 6-6. It can be seen that a considerable amount of the bending strength of the fuselage has been removed by the explosion.

NORTH EAST AIRLINES N8224H DC-6B

During a flight from Philadelphia to Boston on February 24, 1967, North East Airlines N8224H DC-6B suffered explosive decompression due to failure of the right hand forward section of the fuselage in the area of the crew/cargo compartment door. A ten foot section of the fuselage was lost and debris is reported to have caused #3 propeller to break away. The aircraft was landed safely at New York. None of the 9 passengers were injured. At the time of the incident the aircraft was cruising at 15500 feet with cabin differential pressure 5.4 psi. Structural failure initiated from a critical prior cracked condition

American Airlines DC-7 N 316 AA Flight 87 March 5 1957
After explosive decompression due to propeller blade piercing the fuselage.
Aircraft altitude 14000 ft., differential pressure 5.1 psi.

FIGURE 6-5. FLIGHT NO. 87 EXPLOSIVE DECOMPRESSION DAMAGE – THREE-QUARTER REAR VIEW

PHOTO 17

North East Airlines DC-6B N8224H Feb. 24 1967

View of forward fuselage after explosive decompression.

Aircraft altitude 15500 ft., cabin differential pressure 5.4 psi

FIGURE 6-7. EXPLOSIVE DECOMPRESSION DAMAGE

7 CONCLUSIONS

The best locations to place a bomb to do the least damage, in order, are:

- On ventral stairs extended down into slipstream.
- In tail cone as far aft as possible.
- In forward left hand hat rack 20 feet behind the pilot.

An explosion in flight represents a major threat to the safety of the airplane and every effort must be made to minimize its effect. The airplane must be slowed to near the landing speed and pilot maneuvers must be restricted to nearly steady level flight to reduce structural loads. The cabin must be decompressurized to prevent explosive decompression. If it is conceivable that the bomb fuse is pressure sensitive, depressurization should be performed by reducing altitude while maintaining constant cabin pressure. This can be accomplished with automatic settings or with manual override. Even if the bomb is not pressure sensitive, the altitude must be reduced to about 8000 feet if the bomb is left in the cabin because the emergency oxygen system may be destroyed. The oxygen system should be shut off during descent to reduce the fire hazard. Even if the bomb is placed in the tail section, the cabin should remain depressurized for fear that projectiles may rupture the aft pressure bulkhead. People must be kept 20 feet away from the bomb and behind some protection.

An explosion of this size on the ventral stairs would cause minor structural damage; i.e., no primary load paths would be severed. The rudder control would proably be lost. It would be difficult to place the bomb on the aft most step. People would be protected. The airplane could land safely.

An explosion of this size in the tail cone would cause minor damage to the rudder and extensive damage to the fairing structure. No primary load paths would be severed. Rudder control probably would be lost and the elevator control might be lost. It would be easy to place the bomb in the tail cone. People would be protected. The airplane could land safely.

An explosion of this size in the forward fuselage hat rack would cause massive structural damage. The horizontal stabilizer trim control would be lost. The bomb could be placed in the hat rack easily and quickly. The floor acts as a barrier to isolate the blast from the critical equipment. Every effort must be made to protect the floor. The seat backs should be reclined and loose equipment and baggage should be piled on the floor in the region of the direct blast to protect the floor from the initial shock waves. People would be in jeopardy and must seek protection behind obstacles. Calculations show that the airplane could land safely.

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