SEE SUPPLEMENT

Flight Handbook

USAF SERIES
C-119G
AIRCRAFT

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SCOPE . . . This handbook contains the information necessary for safe and efficient operation of the C-119G airplane. These instructions do not teach basic flight principles, but are designed to provide you with a general knowledge of the airplane, its flight characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and elementary instructions have been avoided.

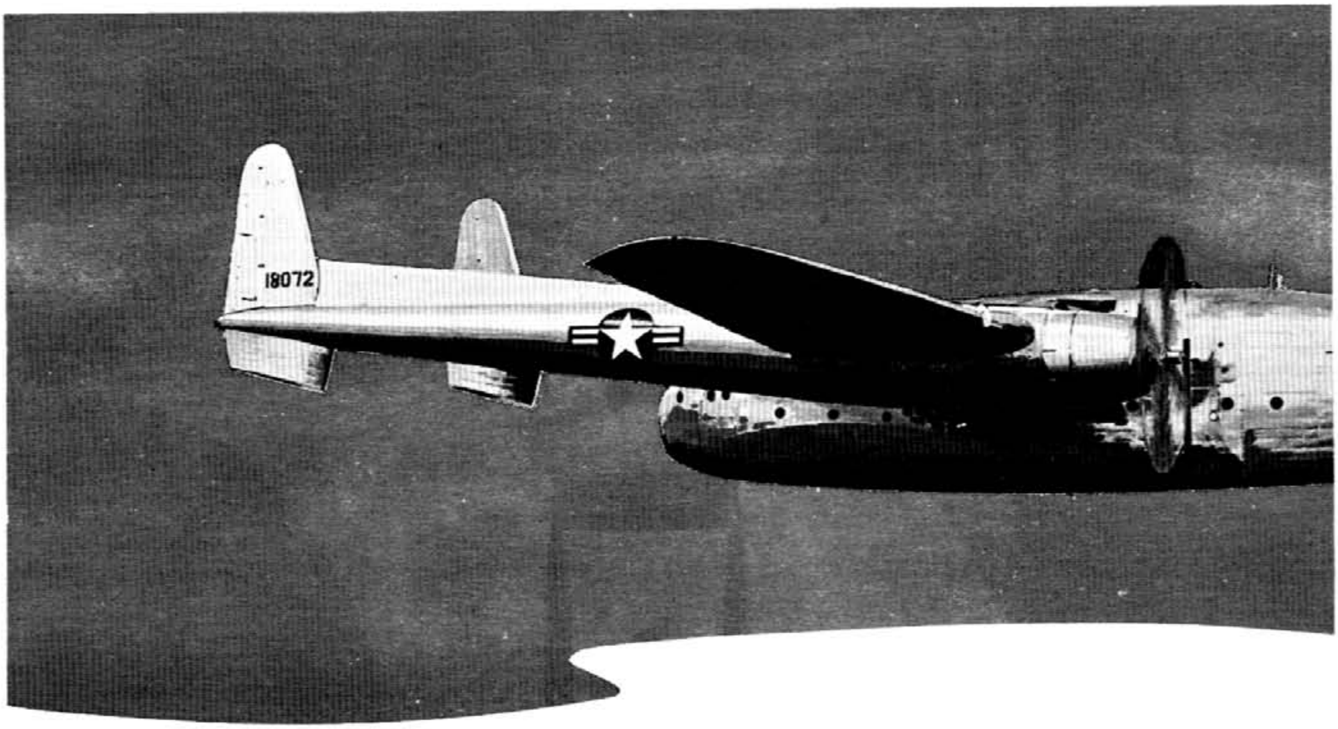
SOUND JUDGMENT . . . The instructions in this handbook are designed to provide for the needs of a crew inexperienced in the operation of this airplane. This book provides the best possible operating instructions under most circumstances, but it is a poor substitute for sound judgment. Multiple emergencies, adverse weather, terrain, etc., may require modification of procedures contained herein.

PERMISSIBLE OPERATIONS . . . The Flight Handbook takes a "positive approach" and normally tells you only what you can do. Any unusual operation or configuration (such as asymmetrical loading) is prohibited unless specifically covered in the Flight Handbook. Clearance must be obtained from ARDC before any questionable operation is attempted which is not specifically covered in the Flight Handbook.

STANDARDIZATION . . . Once you have learned to use one Flight Handbook, you will know how to use them all. Closely-guarded standardization assures that the scope and arrangement of all Flight Handbooks are identical.

ARRANGEMENT . . . The handbook has been divided into 10 fairly independent sections, each with its own table of contents. The objective of this division is to make it easy both to read the book straight through when it is first received and, thereafter, to use it as a reference manual. The independence of these sections also makes it possible for the user to rearrange the book to satisfy his personal taste and requirements. The first 3 sections cover the minimum information required to safely get the airplane into the air and back down again. Before flying any new airplane, these 3 sections must be read thoroughly and fully understood. Section IV covers all equipment not essential to flight but which permits the airplane to perform special functions. Sections V and VI are obvious from their titles. Section VII covers lengthy discussions on any technique or theory of operation which may be applicable to the particular airplane in question. The experienced pilot will probably not need to read this section but he should check it for any possible new information. The contents of the remaining sections are fairly obvious.

YOUR RESPONSIBILITY . . . These Flight Handbooks are constantly maintained current through an extremely active revision program. Frequent conferences with operating personnel and constant review of UR's, accident reports, flight test reports, etc., assure inclusion



of the latest data in the handbooks. In this regard, it is essential that you do your part! If you find anything you don't like about the book, let us know right away. We cannot correct an error whose existence is unknown to us.

PERSONAL COPIES, TABS AND BINDERS . . . Each flight crew member, except those attached to an administrative base, is entitled to have a personal copy of the Flight Handbook. Air Force Regulation 5-13 specifically makes that provision. Flexible, loose leaf tabs and binders have been provided to hold your personal copy of the Flight Handbook. These good-looking simulated-leather binders will make it much easier for you to revise your handbook as well as keep it in good shape. These tabs and binders are secured through your local contracting officer.

HOW TO GET COPIES . . . If you want to be sure of getting your handbooks on time, order them before you need them. Early ordering will assure that enough copies are printed to cover your requirements. Technical Order 0-5-2 explains how to order Flight Handbooks so that you will automatically get all revisions, reissues, and safety of flight supplements. Basically, all you have to do is order the required quantities in the Publications Requirements Table (T.O. 0-3-1). Talk to your base supply officer—it is his job to fulfill your technical order requests. Make sure to establish some system that will rapidly get the books and safety of flight supplements to the flight crews once they are received at the base.

STATUS OF SAFETY OF FLIGHT SUPPLEMENTS . . . You can determine the status of safety of flight supplements by referring to the Index of Technical Publications (T.O. 0-1-1) and the Weekly Supplemental Index (T.O. 0-1-1A). The title page of the Flight Handbook and title block of each safety of flight supplement should also be checked to determine the effect that these publications may have on existing safety of flight supplements.

WARNINGS, CAUTIONS AND NOTES . . . For your information, the following connotation will be applied to these items:

WARNING—Injury to personnel

CAUTION—Damage to equipment

Note—Information requiring emphasis

COMMENTS AND QUESTIONS . . . Comments and questions regarding any phase of the Flight Handbook program are invited and should be addressed to Detachment No. 1, Air Research and Development Command, Wright-Patterson Air Force Base, Ohio, Attention: RDZSTH.

Figure 1-1. The Airplane

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THE AIRPLANE.

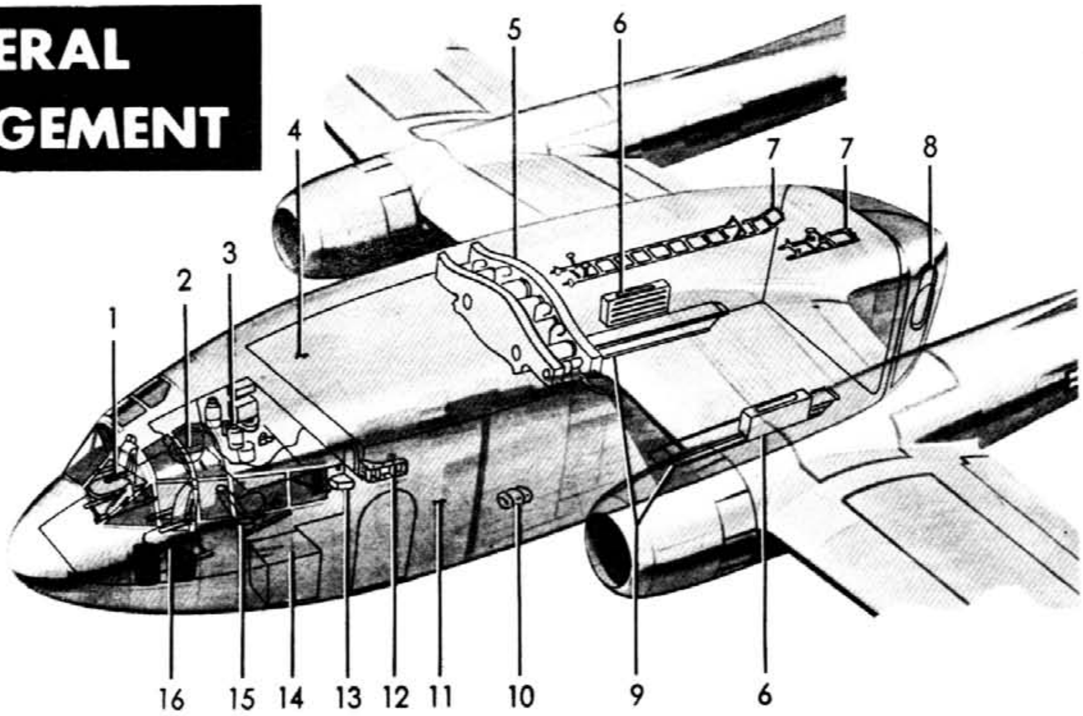
Built by Fairchild Aircraft Division, the C-119G airplane is a two-engine, twin boom, high wing, land-type monoplane. It features a tricycle type landing gear with dual wheels on each gear strut. Constant speed, four-bladed, reversible pitch propellers are driven by two supercharged radial engines. It was designed for and is being used as (1) a cargo carrier, (2) a troop or paratroop transport, (3) an ambulance evacuation airplane, (4) as a cargo drop airplane with facilities for aerial delivery of both heavy and light equipment and supplies.

SIZE.

Wing span	109 feet 3-1/4 inches
Fuselage length	60 feet 6-3/8 inches
Overall length	86 feet 5-3/4 inches
Height	26 feet 2-5/8 inches

The contour of the airplane fuselage is such that a rectangular cross section exists throughout the main cargo hold, permitting easy stowage of numerous types of cargo. The tricycle landing gear positions the entire cargo compartment in a level attitude when the airplane is parked. The height of the floor above the

GENERAL ARRANGEMENT



1. COPILOT'S STATION
2. NAVIGATOR'S STATION
3. LAVATORY COMPARTMENT
4. AUXILIARY EQUIPMENT COMPARTMENT
5. HEATER COMPARTMENT
6. STOWAGE LOCKER
7. MAINTENANCE LADDER STOWAGE
8. DUST EXCLUDER STOWAGE
9. CARGO LOADING RAMP STOWAGE
10. ANTENNA KIT
11. CARGO COMPARTMENT
12. MAIN ENTRANCE LADDER STOWAGE
13. FLIGHT MECHANIC'S STATION
14. BAIL-OUT CHUTE
15. RADIO OPERATOR'S STATION
16. PILOT'S STATION

Figure 1-2

ground is approximately four feet, or truck-bed level. The cargo compartment dimensions are:

Height....................................8 feet
Width...........................9 feet 2 inches
Length........................36 feet 11 inches
Area of Cargo Floor..................353 square feet

GROSS WEIGHT.

The design weight of the airplane is 64,000 pounds. Refer to Section V for additional gross weight limits.

CREW.

The crew normally consists of pilot, copilot, radio operator, navigator and flight mechanic.

FUNCTIONS.

AS A CARGO CARRIER.

Under various loading conditions, the airplane will transport 75-mm howitzers, 37-mm guns, 40-mm anti-aircraft guns and carriages, 10-wheel 2-1/2 ton trucks, large and small aircraft engines and cradles, propellers, and a wide variety of other military equipment. When equipped with special ramps and load-distributing devices, it can also carry 75-mm guns and halftracks and 155-mm howitzers. Close attention should be paid to proper securing methods, as well as weight distribution.

AS AN AMBULANCE EVACUATION AIRPLANE.

As an ambulance, the airplane carries 35 litters, 20 on the right and 15 on the left, arranged along each side of the cargo compartment. The litters are mounted in seven tiers, 5 litters high, and each is supported by posts and web straps. Thirty-five is the maximum number of litter patients which may be carried, but various combinations of troops or seated casualties and litter patients may be obtained by removing the lower level litters which will allow the use of seats. A total of 76 troops may be carried when the airplane is used for emergency evacuation purposes, with 62 troops seated and 14 litter patients.

AS AN EQUIPMENT DROP AIRPLANE.

An electrically operated, automatic aerial delivery system is employed to drop equipment or supplies through the paratainer doors and is capable of dropping twenty 500-lb. bundles in 8 to 10 seconds. For dropping heavy, bulky equipment, the rear clamshell doors may be removed or the flight operable door may be used on airplanes having the flight operable door feature. Heavy equipment is extracted through the open aft end of the fuselage and lowered by a series of parachutes.

AS A TROOP TRANSPORT.

The airplane, with 20 folding seats along the left side of the fuselage, and 22 seats along the right side, carries 42 troops or paratroops. In addition, provisions are made for installation of 20 more troop seats along the center of the cargo compartment floor, giving a total troop carrying capacity of 62.

ENGINES.

MODEL R-3350-85 ENGINES.

The Model R-3350-85 engine, two of which supply power for the airplane, is an air-cooled, compound, radial, reciprocating power plant. The engine incorporates three blow-down turbines located 120 degrees around the circumference of the engine. Although similar in construction to a turbo-supercharger, these turbines utilize kinetic energy rather than the pressure of the engine exhaust; and instead of driving a supercharger that rams the tops of the carburetor, they are geared directly to the crankshaft through a system of Zerol beveled gears. A fluid coupling effects the transfer of power to the crankshaft drive gear. There are no separate controls for these turbines as they run at a constant ratio of 7.83-1 to crankshaft speed. Thus, when the engine develops its normal rated power at 2600 rpm, the turbines turn at a speed of 20,370 rpm. The power derived from these turbines is approximately 150 bhp per turbine when the engine is operated at take-off power. The turbines are designed to operate efficiently at all altitudes. They do not require large pressure differentials in the exhaust system, thus obviating undesirable high back pressures in the cylinders. By supplementing energy derived from engine fuel, the turbines permit higher horsepower at lower fuel consumption. The engine is equipped with a two-speed supercharger, impeller injection system, fuel metering system, water injection system, torquemeter, and low tension ignition system. At take-off, each engine will produce 3250 bhp at 2900 rpm without water injection and 3500 bhp at 2900 rpm with water injection.

MODEL R-3350-89 ENGINES.

When the R3350-85 engine is modified either in production or at the time of overhaul to incorporate a low-ratio turbine drive, drilled turbine wheels, and armored exhaust hoods, the engine is redesignated as the R3350-89. Engine performance is unaffected and the same engine limitations apply. The low-ratio turbine drive runs at a constant ratio of 6.52-1 to crankshaft speed; at normal power, the turbines turn at a speed of 16,950 rpm.

MAIN DIFFERENCE TABLE

	C–119B	C–119C	C–119F	C–119G
ENGINES	R–4360–20	R–4360–20WA	R–3350–85, –89, –89A	R–3350–85, –89, –89A
WATER INJECTION	No	Yes	Yes	Yes
HORIZONTAL STABILIZER TIP	Yes	49–119 thru 49-199	No	No
LANDING GEAR	Electric	Electric 49—119 thru 51-2584 Hydraulic 51–2587 thru 51–8273	Hydraulic	Hydraulic
WING FLAPS	Electric	Electric 49–119 thru 51–2584 Hydraulic 51–2587 thru 51–8273	Hydraulic	Hydraulic
PROPELLERS	Ham. Stan.	Ham. Stan.	Ham. Stan.	Aeroproducts

Figure 1–3

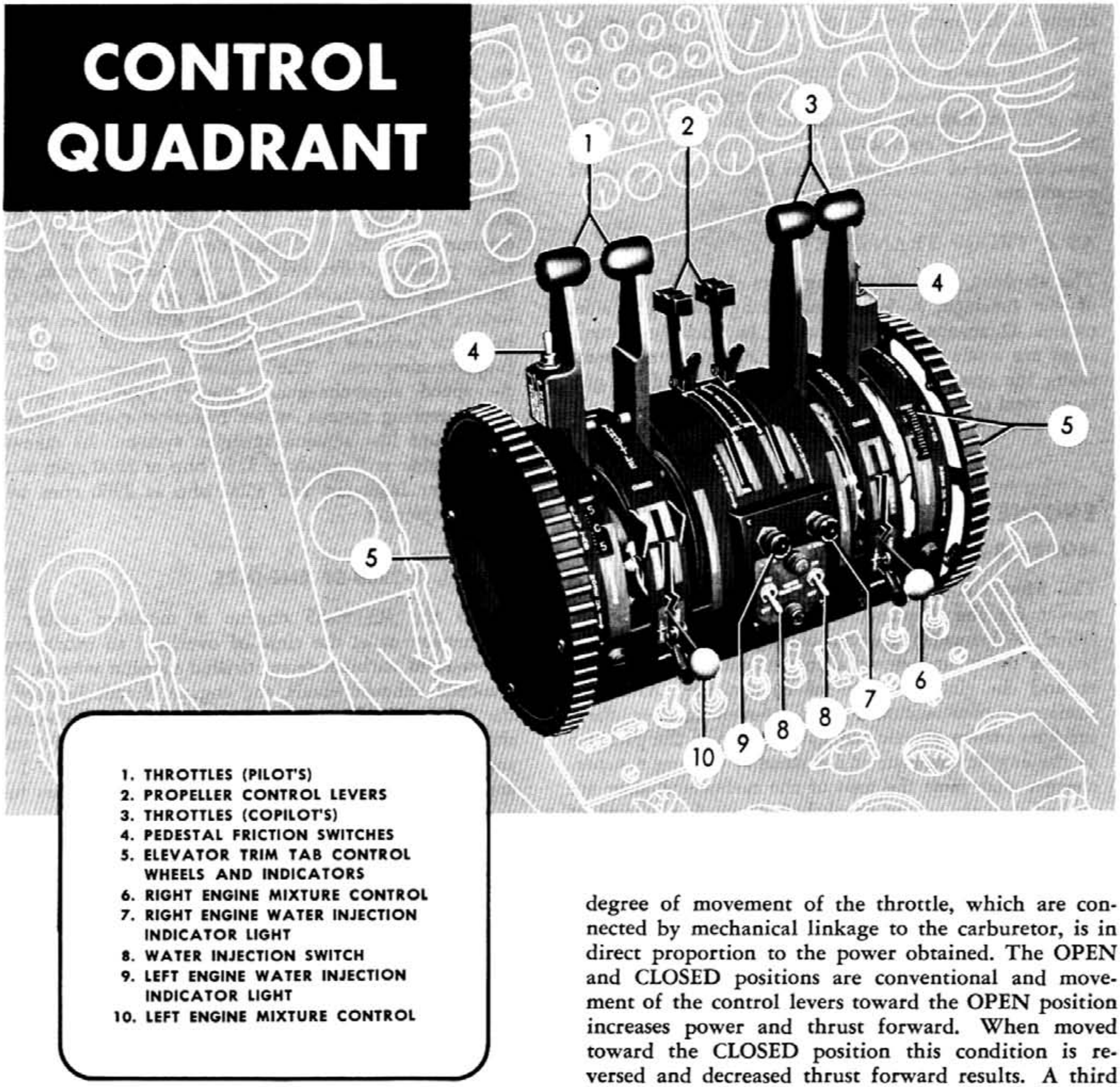


Figure 1-4

MODEL R-3350-89A ENGINES.

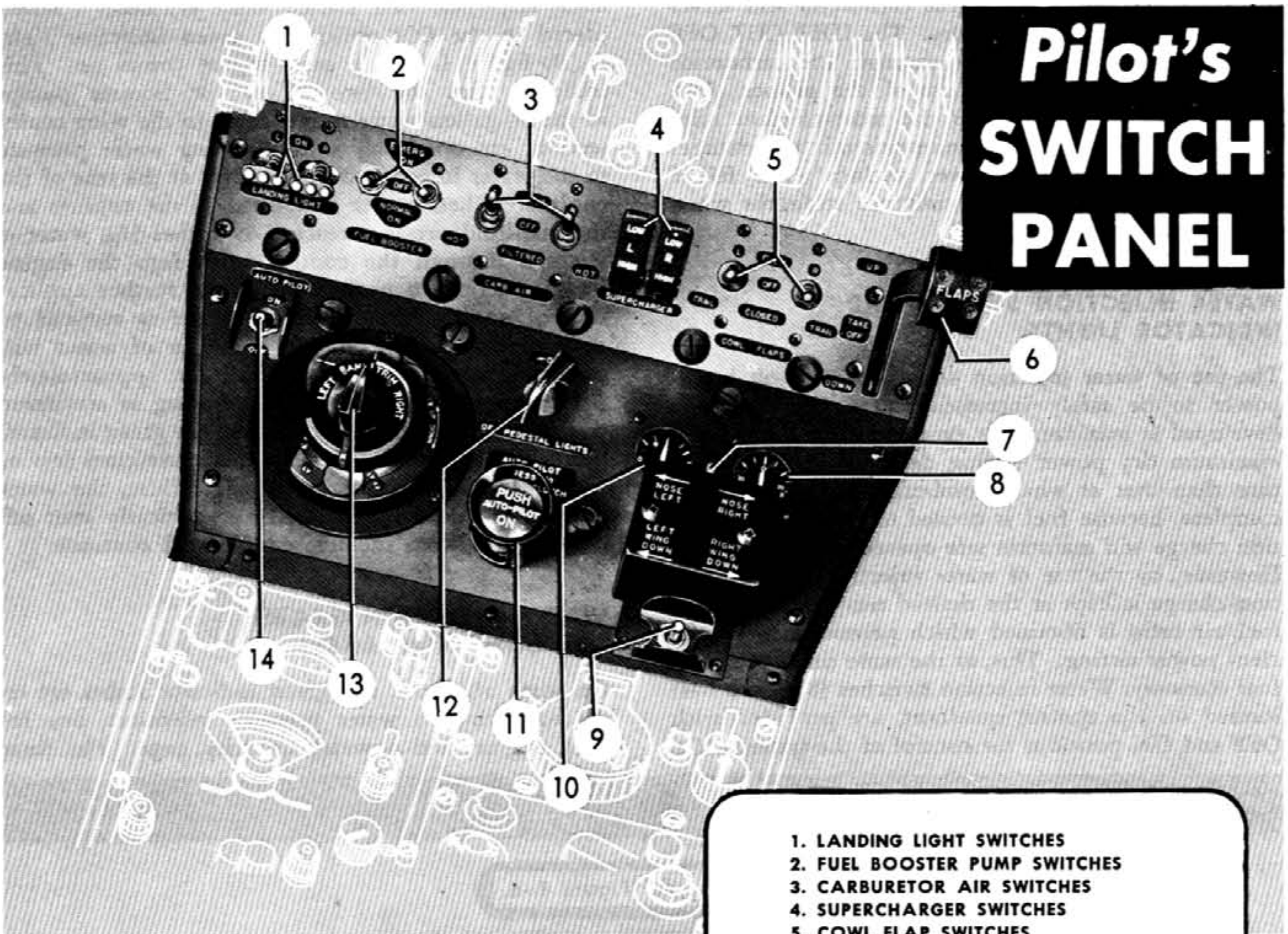
When the -89 engine is further modified to incorporate an improved torquemeter system, a four-pinion cam drive, and HC-250 valve guides, the engine is redesignated the -89A. Engine performance is unaffected and the same engine limitations apply.

THROTTLES.

Throttle levers (1, 3, figure 1–4) are installed in the control quadrant. The system is so designed that the degree of movement of the throttle, which are connected by mechanical linkage to the carburetor, is in direct proportion to the power obtained. The OPEN and CLOSED positions are conventional and movement of the control levers toward the OPEN position increases power and thrust forward. When moved toward the CLOSED position this condition is reversed and decreased thrust forward results. A third position, REVERSE, is obtained by lifting the throttles after the CLOSED position has been reached, and continuing the aft motion.

Further reverse movement of throttle controls increases in proportion the reverse power. Movement of the controls to the extreme aft position will allow 80% of maximum continuous power to be applied for reversing operations. When the left throttle is moved into the reverse range, the action electrically (28-volt dc) positions a hydraulic selector valve, thus directing the flow of hydraulic oil to the elevator locking plates. The elevator is then locked and movement of this surface is prevented during taxiing and reversing operations. The elevator will remain locked until the throttle is moved into the forward range to a manifold pressure of approximately 38 inches Hg.

Pilot's SWITCH PANEL



1. LANDING LIGHT SWITCHES
2. FUEL BOOSTER PUMP SWITCHES
3. CARBURETOR AIR SWITCHES
4. SUPERCHARGER SWITCHES
5. COWL FLAP SWITCHES
6. WING FLAP LEVER
7. RUDDER TRIM TAB SWITCH
8. RUDDER TRIM TAB INDICATOR
9. AILERON TRIM TAB SWITCH
10. AILERON TRIM TAB INDICATOR
11. AUTOMATIC PILOT SERVO CLUTCH SWITCH
12. PEDESTAL LIGHTS RHEOSTAT
13. AUTOMATIC PILOT TURN-AND-PITCH CONTROLLER
14. AUTOMATIC PILOT POWER SWITCH

Figure 1-5

THROTTLE REVERSE STOPS.

Incorporated into the pilot's throttle system are a set of movable stops which, during flight, make contact with the pilot's throttles and prevent inadvertent reversal of the propellers. The throttle stops consist of an enclosed track along the top of the quadrant with a 90° offset of approximately 0.5 inch. The throttles contact the stops at the CLOSED or idle position and must be lifted in order to enter the REVERSE THRUST range. When either of the pilot's throttles is lifted (and the main landing gear struts are compressed), the associated throttle lift switch installed on the rear of the quadrant closes an electrical circuit energizing a 28-volt dc actuator which retracts the throttle stop and permits aft movement of the throttle into the REVERSE THRUST range. Actuation of the throttle lift switch also causes the propeller reversing coil to be energized. The throttle reverse stop circuit is routed through the main landing gear strut switches to prevent the use of reverse thrust until the airplane is on the ground.

PEDESTAL FRICTION SWITCHES.

A pedestal friction switch (4, figure 1–4) is located on each set of throttles so that either the pilot or copilot may change the amount of friction necessary to hold the throttles in a desired position and keep them from creeping. When the friction switches are toggled to INCREASE or DECREASE, they control a 28-volt dc actuator in the forward end of the pedestal which, through mechanical linkage, adjusts the friction device. Either pedestal friction switch controls the friction of both sets of throttles.

MIXTURE CONTROLS.

Mixture control levers (6, 10, figure 1–4) are provided on the control quadrant. Each of these controls is connected through mechanical linkage to the corresponding left and right engine carburetor and serves to control the fuel/air mixture. The IDLE CUT OFF position stops all fuel flow through the carburetor except that resulting from use of the primer. The NORMAL position provides the most efficient mixture of fuel/air, providing automatic mixture changes as required by various power settings. The RICH position provides the richer mixture strengths when necessary during operation.

WATER INJECTION SWITCH AND INDICATOR LIGHTS.

The use of water injection allows the take-off horsepower to be safely increased above that normally developed at maximum dry power settings. The usual maximum dry power setting demands a richer mixture than that at which combustion would be most complete, because fuel is needed to cool the combustion charge and maintain safe operating temperatures. However, by the use of water injection, the combustion charge is kept at the desired temperatures and a fuel/air ratio is obtained which permits a more efficient power setting. This, at the same time, promotes fuel economy. Water injection switches (8, figure 1–4) located on the control quadrant, are provided with OFF and ON positions for control of the system. (On later airplanes, one switch only is used.) When placed in the ON position, a green indicator light (7, 9, figure 1–4), for each engine comes on. This indicates that the two 28-volt dc booster pumps in the 56-gallon water tank located in the wing center section, are operating and supplying water pressure to the water injection control valves at the rear of the engines. When power is applied to the engines and the manifold pressure reaches 45 inches Hg, water is metered through the control valve into the engine blower section in proportion to the further application of engine power. When power is reduced to approximately 46-47 in. Hg, the flow of water will cease. A full tank of water will last approximately 8 minutes with both engines operating at maximum power. Should this supply run out, the green indicator lights will go out and indicate the interrupted flow of water. In the event of an engine failure, the water flow to the dead engine will automatically shut off, while the flow to the live engine will continue.

WATER INJECTION GAGES (ON SOME AIRPLANES).

On some airplanes, the green indicator lights are replaced with two water pressure indicators, one for each engine, and a water quantity gage. The three

Legend for Figure 1–6.

1. INDIVIDUAL HEATER SWITCHES
2. HEATER STARTER BUTTON
3. HEATER MASTER SWITCH
4. HEATER INDICATOR LIGHTS
5. PITOT HEATER SWITCH
6. WINDSHIELD ANTI-ICING SWITCH
7. WING AND TAIL ANTI-ICING SWITCH
8. RADOME ANTI-ICING SWITCH
9. RADIO CONTROL PANEL SPOT LIGHT
10. BATTERY SWITCH
11. MAGNETIC COMPASS LIGHT RHEOSTAT
12. AERIAL DELIVERY SALVO SWITCH
13. PASSING LIGHT SWITCH
14. NAVIGATION LIGHTS CONTROL SWITCH
15. NAVIGATION LIGHTS DIM-BRIGHT SWITCH
16. FORMATION LIGHTS SWITCH
17. MAIN AND SPARE INVERTER SWITCH
18. AUTO PILOT INVERTER SWITCH
19. INSTRUMENT INVERTER SWITCH
20. OIL COOLER EXIT FLAP SWITCHES
21. RIGHT ENGINE FUEL SELECTOR SWITCH
22. RIGHT ENGINE FUEL, OIL AND HYDRAULIC SHUT-OFF SWITCH
23. RIGHT ENGINE FIRE DETECTOR LIGHT
24. RIGHT ENGINE FIRE EXTINGUISHER SWITCH
25. HEATER FIRE EXTINGUISHER SWITCH
26. FIRE DETECTOR TEST BUTTON
27. HEATER FIRE DETECTOR LIGHT
28. AUXILIARY POWER PLANT FIRE DETECTOR LIGHT
29. LEFT ENGINE FIRE EXTINGUISHER SWITCH
30. LEFT ENGINE FIRE DETECTOR LIGHT
31. LEFT ENGINE FUEL, OIL AND HYDRAULIC SHUT-OFF SWITCH
32. LEFT ENGINE FUEL SELECTOR SWITCH
33. GENERATOR SWITCHES
34. STARTER SWITCH
35. PRIMER SWITCH

35A. HOT PRIME SWITCH

36. IGNITION SWITCHES
37. OVERHEAD PANEL LIGHT RHEOSTAT
38. OIL DILUTION SWITCHES
39. COCKPIT RED DOME LIGHT RHEOSTAT
40. COCKPIT CLEAR DOME LIGHT RHEOSTAT
41. PROPELLER DE-ICING SWITCH
42. CARGO COMPARTMENT HEAT MASTER SWITCH
43. RADIO CONTROL PANEL SPOTLIGHT
44. ENGINE ACCESSORY HEAT SWITCH
45. COCKPIT AIR SWITCH

Overhead PANEL

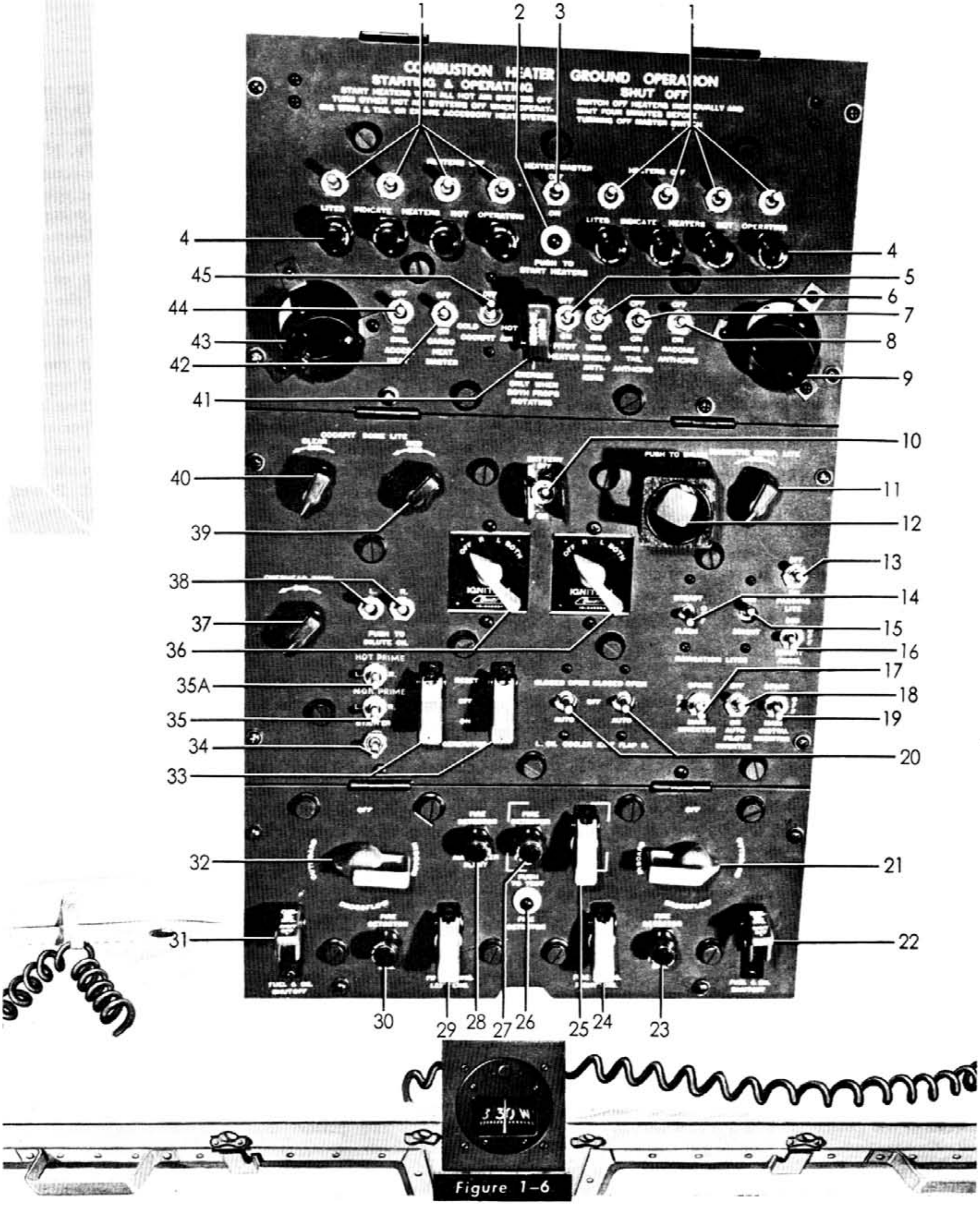


Figure 1-6

gages are located just below the fuel quantity gages on the center instrument panel. The water quantity gage, operating on 28 volts dc, provides an indication of the amount of water-alcohol available in the supply tank. The 26-volt ac water pressure gages indicate not only the water pressure available for use but the correct flow of water to the engine. Refer to Instrument Range Markings, Section V.

SUPERCHARGER SWITCHES.

Two supercharger switches (4, figure 1–5), one for each engine, are installed on the pilot's switch panel on the pedestal to control the two-speed, single-stage supercharger incorporated into each engine. The shift in blower speeds is electrically controlled by the switches. During normal operation and starting, the supercharger switches are placed in LOW. The HIGH position of the switch electrically operates a 28-volt dc solenoid valve and actuator unit located on the aft end of the engine accessory section. The solenoid valve and actuator unit control the oil flow to the supercharger clutch assembly which, then, causes the supercharger mechanism to shift into high ratio. Refer to Section V for supercharger limitations.

IGNITION SWITCHES.

Two ignition switches (36, figure 1–6) are utilized for engine ignition purposes only. These switches remain in BOTH during normal airplane operation, except during magneto check, at which time they are switched momentarily to the L or R position. When the switch is OFF, the system is inoperative.

STARTER SWITCH.

A single switch (34, figure 1–6), spring loaded to the center OFF position, is located on the overhead panel. When this switch is held in either the L or R position it electrically energizes the starter circuit allowing 28-volt dc current to flow from the nacelle junction box to the direct cranking starter of the left or right engine. Simultaneously the output of an induction vibrator is added to the engine ignition system to boost the spark voltage of the front spark plugs of all cylinders. The boost circuit is disconnected when the starter switch is released.

COWL FLAP SWITCHES.

The cowl flaps of each engine are controlled by cowl flap switches (5, figure 1–5) on the pilot's switch panel, each having an OPEN, CLOSED, TRAIL and OFF position. When either switch is moved to a selected position, the corresponding 28-volt dc cowl flaps power unit is energized. The power unit is connected to the geared ends of a segmented flexible shaft which interconnects the nine screwjacks and, when energized, simultaneously positions the nine magnesium cowl flaps. The OPEN and CLOSED positions are conventional; the TRAIL setting positions the cowl flaps approximately 3° from the closed position. Intermediate positioning of the flaps is accomplished by toggling the switch to the OPEN or CLOSED position and then to OFF.

CARBURETOR AIR SWITCHES.

The left and right carburetor air systems are separately controlled by a four position switch (3, figure 1–5) provided for each system. The positions of the switches are COLD, HOT, FILTERED and OFF. When placed in the HOT position, the mixing valve actuator is energized and regulates the temperature of the air by restricting the flow of cold air from the wing air scoop and permitting the hot air, which is heated in its passage over the exhaust system, to flow into the mixing valve box. In the COLD position of the switch, the hot air is similarly restricted and the cold air is permitted to flow into the mixing valve box. To vary the temperature between full hot or full cold conditions, the switches may be momentarily moved to either HOT or COLD so that the actuator is stopped at some point other than a limit of travel. The FILTERED position of the switch energizes the filter valve actuator which directs air through a filter in the cold air duct. The center OFF position of the switch will hold the last carburetor air condition selected. Power for the operation of the carburetor air systems is derived from the 28-volt dc electrical system.

ENGINE PRIMER SWITCH.

A 28-volt dc fuel priming system is provided for each engine and is controlled by a spring-loaded switch (35, figure 1–6) located on the overhead panel. However, to obtain fuel pressure for this operation, the booster pumps must be operating to force the fuel to the carburetor. The primer switch electrically controls a solenoid valve which is an integral part of the carburetor and allows the fuel under pressure to be forced from the unmetered chamber of the carburetor, through the primer line, into the supercharger rear housing. The three switch positions are L and R for the left and right engine and the center position, which is OFF.

HOT PRIME SWITCH.

On airplanes modified in accordance with T.O.1C-119-549, a hot fuel priming system is installed to facilitate cold weather engine starts. Complete operation of the 28-volt dc hot prime system is controlled by a three-position switch (35A, figure 1–6) on the overhead panel. The placarded switch positions are L and R; the spring-loaded OFF position is unmarked. When, for example, the switch is held in the R position, the primer solenoid valve on the right engine carburetor is opened, the automatically-regulated fuel heating unit is energized and heated fuel is injected into the supercharger rear housing.

A similar sequence of left engine priming occurs when the hot prime switch is held in the L position. The hot prime and normal prime fuel lines and primer solenoid valves are common to both systems; the only difference in the operation of the systems is the energization of the heating units when the hot prime switch is used. As in normal prime operation, the fuel booster pumps must be operating to supply pressure for hot prime operation. Refer to Section VII for a more detailed description of the hot fuel priming system.

HOT PRIME TEMPERATURE GAGES.

A 28-volt dc temperature gage (12, figure 1-22) for the hot prime installation on each engine is located on the instrument panel. These gages, which are calibrated in degrees Centigrade, indicate the temperature of the heated fuel and provide a check of system operation. Refer to Instrument Range Markings, Figure 5-1, for the temperature gage marking.

TACHOMETERS.

Right and left engine tachometers (49, figure 1-22) located on the instrument panel, indicate the rpm developed by the respective engine crankshaft. Power individually developed by the tachometer generators is used to drive the indicators.

CYLINDER HEAD TEMPERATURE INDICATORS.

Two cylinder head temperature indicators (40, figure 1-22) are located on the instrument panel and indicate the cylinder head temperature in degrees Centigrade. The signal which operates these indicators is generated by a sensing element which is installed in the number one cylinder of each engine.

CARBURETOR AIR TEMPERATURE INDICATORS.

A bulb-thermometer, installed at the mixing valve assembly above the carburetor, senses and transmits the temperature of the air entering the carburetor to the carburetor air temperature indicators (41, figure 1-22). There it is indicated in degrees centigrade. The carburetor air temperature indicators operate on 28-volt dc.

TORQUEMETERS.

The torquemeter (48, figure 1-22) located on the instrument panel indicates in psi (pounds per square inch) the actual power delivered by the propeller shaft. The signal which the torquemeter indicators register is transmitted electrically from the 26-volt ac torquemeter transmitter installed in the engine. This information is used to obtain an accurate determination of true engine power delivery. Refer to Section VII for calculation of brake horsepower.

MANIFOLD PRESSURE GAGES AND PURGE VALVES.

The manifold pressure gages (50, figure 1-22) are used to indicate the pressure of the fuel/air mixture in the engine intake system as controlled by the throttles. This information is needed to obtain desired power settings and to prevent overboosting. In addition it serves as a guide for adjusting the supercharger to the need of the engine at low or high altitudes. Pressure is taken from the engine blower section and used to operate a diaphragm in the instrument. This movement is directly responsible for the movement of the indicator pointer. Two manifold purge valve buttons (10, figure 1-22), located on the instrument panel, provide a means of removing moisture and foreign matter from the manfold pressure gage lines. At idling speeds the pressure in the manifold is less than atmospheric pressure and depressing the button allows the moisture to be drawn into the engine.

Note

Do not operate purge valves except when engines are idling. Only when the manifold pressure is less than atmospheric pressure, will the moisture or foreign matter in the lines be drawn into the engine.

PROPELLERS.

Each engine drives an Aeroproducts A644FN-C2 propeller which is a four bladed, self-contained, hydraulically controlled, constant speed propeller incorporating full feathering and reverse pitch operation and electric blade de-icing. Each propeller has an integral oil system with a 9-10 quart capacity.

Note

Although some airplanes are equipped with Aeroproducts A644FN-C1 propellers, modification of these propellers to conform to C2 propellers is being accomplished by compliance with T.O. 1C-119G-507. Operation and procedures with A644FN-C1 propellers (which differ in several specific instances from those outlined in this handbook) are adequately covered by existing safety-of-flight supplements against this handbook.

Propeller forward thrust operation is controlled by the propeller control levers on the control quadrant, each of which is mechanically linked by cable and teleflex rigging to a control ring on its associated regulator. When the control lever is placed full forward in the propeller governing range, the propeller blades assume a low pitch-high rpm position. As the control levers are returned to some position in the propeller governing range, the propeller blade angle is determined by the control lever setting and the power at which the engine is operating. The high pitch limit of the forward thrust governing range is fixed by a detent which engages a safety latch on each control lever.

Synchronization of the propellers is accomplished by manual setting of the propeller control levers in conjunction with the tachometers on the instrument panel. Further adjustment of the propellers to eliminate noise beats must be done aurally. When the propeller control levers are placed full forward, both engine speeds are aligned to the same setting, as the propeller regulators are then set alike for maximum rpm. However, because of permissible tolerances in propeller regulator settings, the control levers at this maximum rpm setting will only set the speed of the engine within a few rpms of each other. This slight difference is enough to cause propeller noise beats normally referred to as "out-of-synchronization."

Either propeller is feathered by releasing the safety latch on its associated control lever, and moving the lever to the full extent of its rearward travel. Feathering is completed in approximately 8 to 10 seconds. Unfeathering is accomplished by returning the control to a full low pitch position in the forward thrust governing range. Momentary overspeeding of a cold engine during the unfeathering operation is reduced by setting the control lever to the mid point of the governing range until engine temperatures stabilize.

Reverse thrust for breaking the landing roll of the airplane is obtained by lifting the pilot's throttles and moving them rearward beyond the CLOSED or idling position. Lifting the throttles not only causes the quadrant reverse stops to retract but also energizes the propeller reversing coils which release the positive electro-mechanical low pitch stops on the propeller master gear assemblies. As the pilot's throttles are moved into the REVERSE THRUST range, an interconnecting linkage in the engine throttle control system positions the negative thrust control ring at the propeller regulator which results in a constant blade angle of −15.5 degrees. Further rearward movement of the throttles advances engine power and reverse thrust. The propeller, however, does not govern in reverse pitch and engine speed is manually controlled by positioning the throttles for desired engine power. The maximum obtainable engine power for reverse thrust is approximately 80% of normal rated power. Unreversing is accomplished by returning the throttles to a position in the forward thrust range.

PROPELLER CONTROL LEVERS.

Two propeller control levers (4, figure 1–4), one for each propeller, are installed on the center of the control quadrant and provide the means for the control of the propeller in forward thrust and feathering. Each control has an INCREASE RPM position (forward), a DECREASE RPM position (aft), and a FEATHER position (extreme aft). A spring-loaded safety catch is incorporated in each control lever which engages a detent at the high pitch-low rpm end of the propeller governing range. This catch prevents inadvertent movement of the propeller control levers into the FEATHER position.

OIL SYSTEM.

An individual pressure-type oil system with dry crankcase and external oil tank lubricates each engine. A self-sealing oil tank having a capacity of 60 gallons, plus an expansion space of 12 gallons, is located in the top portion of each of the two nacelles just forward of the wheel wells. Oil specification and grade are given in Servicing Diagram (Figure 1–25). An oil cooler provided for each engine is located in the leading edge of the wing outboard of the nacelle and forward of the front spar. Cooling air is ducted to the cooler from a scoop in the wing leading edge. Air flow is regulated by shutters mounted downstream of the oil cooler, and a thermostat installed on the engine oil supply line provides automatic temperature regulation through control of the exit shutters. On later airplanes a check valve is installed in the oil return line between the oil cooler and the engine to prevent oil from leaking from the cooling system back to the engines after shutdown. An oil dilution system provided for each engine allows fuel to flow into the oil system, thinning the lubricating fluid before the power plant is shut down. A 30% dilution of the oil system may thus be accomplished in four minutes. Electrically operated oil shut-off valves are installed aft of the firewall to permit stoppage of oil flow to the engine. Indicators are installed to give the pilot an indication of the following conditions prevailing in the system: oil pressure, oil quantity, and oil temperature.

OIL COOLER EXIT FLAPS SWITCHES.

Two oil cooler exit flaps switches (20, figure 1–6), located on the overhead panel, control the flow of air through the oil cooler by regulating the shutter on the exit of the coolers, thereby keeping the oil which is returning from the engine at a desired temperature before it re-enters the oil tanks, and subsequently the engines. During normal operation, the switches are in the AUTO position; however, if system malfunction should make automatic operation impossible, the exit flap may be manually operated by the OPEN and CLOSED position of the switches. Automatic operation of the oil cooler exit flaps is maintained by the oil cooler floating control located in the engine oil supply line immediately aft of the firewall. The floating control automatically regulates the oil temperature by electrically positioning the oil cooler exit flaps through 28-volt dc actuators according to the temperature demand. When placed in the spring-loaded OPEN or CLOSED positions, electrical power bypasses the floating control and directly energizes the exit flaps actuators, thus setting the flaps according to the switch selection. The center OFF position of the switch stops the flaps at their last position.

OIL TEMPERATURE INDICATORS.

Right and left engine oil temperature indicators (43, figure 1–22), located on the instrument panel, record the temperature in degrees C of the oil supply as transmitted by sensing elements attached to the oil inlet firewall casting on the aft left side of the firewall and extending into the oil flow. Power to operate the oil temperature indication system is obtained from the 28-volt dc electrical system.

OIL DILUTION SWITCHES.

Two oil dilution switches (38, figure 1–6), located on the overhead panel and spring-loaded to the OFF position, are utilized during cold weather to thin the engine oil with fuel before shut down. Fuel for oil dilution is supplied from the unmetered fuel chamber in the carburetor through a 28-volt dc solenoid valve directly to the oil system line at the firewall fitting.

FUEL, OIL AND HYDRAULIC SHUT-OFF SWITCHES.

A fuel, oil, and hydraulic oil shut-off switch (22, 31, figure 1–6) is provided for each engine. Located on the overhead panel, these switches remain in NORMAL position during normal operation with the switch down and switch guard closed. In an emergency such as an engine fire, the corresponding switch is placed in the SHUT position, with the switch up and the switch guard open. This closes the 28-volt dc shut-off valves at the firewall, causing the flow of all fuel, engine oil and hydraulic oil to be shut off from the affected engine.

OIL PRESSURE INDICATORS.

Right and left engine oil pressure indicators (42, figure 1–22) are located on the instrument panel to indicate the oil pressure developed by the respective engine. The inverter, which supplies power to the 26-volt ac transformer, must be operating to obtain oil pressure indication.

OIL QUANTITY INDICATORS.

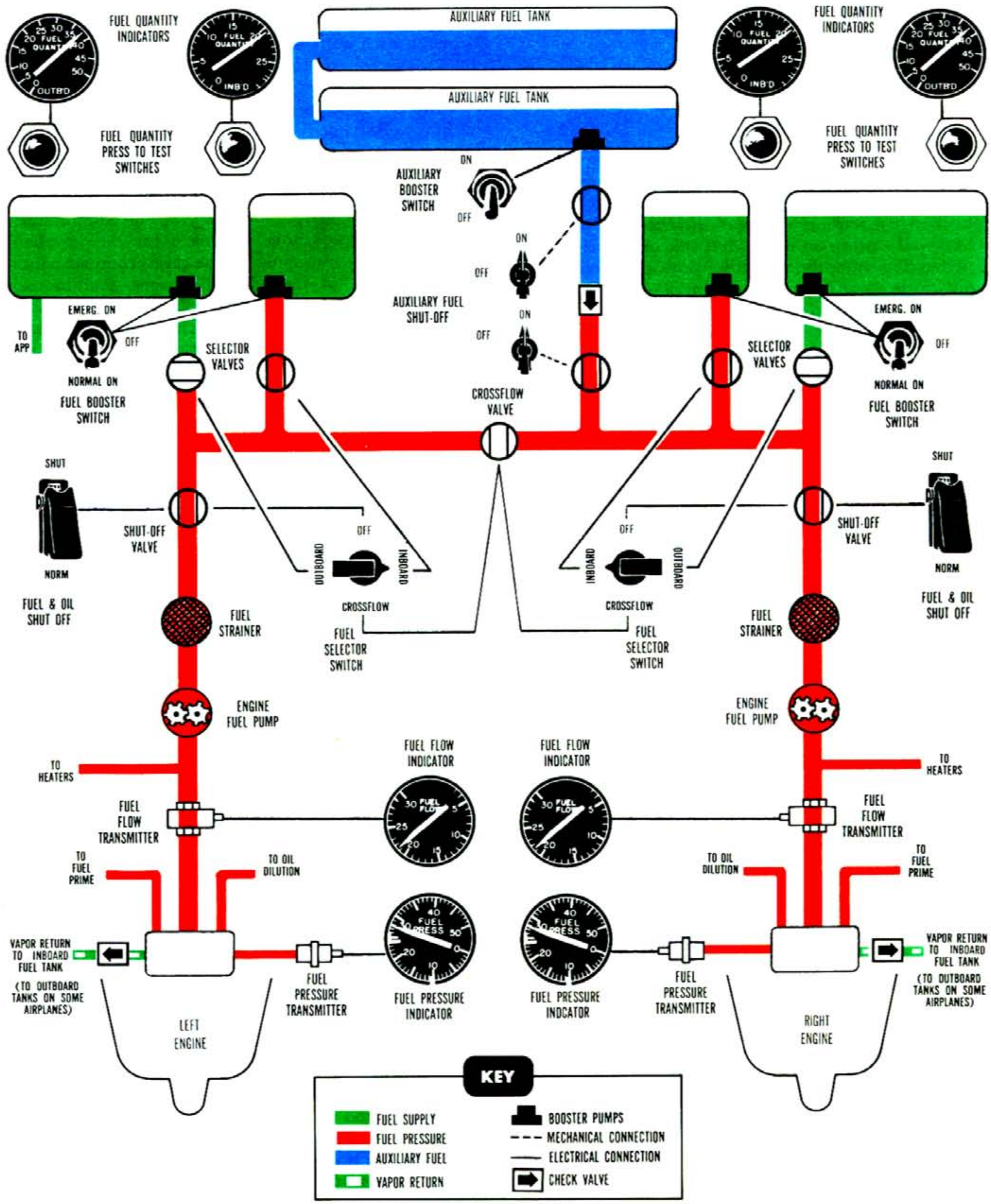
The oil quantity as transmitted by the 28-volt dc transmitter in each oil tank is indicated by the respective oil quantity indicator on the instrument panel (44, figure 1–22).

FUEL SYSTEM.

Although each engine is provided with its own separate fuel system and fuel supply tanks, the two systems are interconnected by a crossflow system which makes it possible to operate either engine from any one of the four tanks. Facilities for engine priming, oil dilution, and vapor return from the carburetor are incorporated into each system, as are shut-off switches for preventing fuel flow to the engines should an emergency condition arise. With a view to long range ferry missions, provisions are made for the installation of auxiliary fuel tanks in the cargo compartment. Four fuel tanks of the self-sealing type contain the normal fuel supply. One tank is located in each side of the center section and one in each wing outer panel. Each set of tanks is interconnected and by use of selector switches in the crew compartment may be used to supply fuel to either engine. Fuel for the operation of the auxiliary power plant system is taken directly from the number 5 cell of the left outboard tank. Heater fuel supply is obtained from the engine-driven fuel pumps when the engines are operating; with engines inoperative, booster pumps must be used to supply the fuel under pressure.

The highest cell of each tank is vented to the atmosphere in such a manner that no spilling or siphoning of the fuel is possible. Each tank is provided with a marked filler neck and drain for overflow fuel. A sump is located at the bottom of each tank with a drain to permit removal of accumulated water and foreign matter. The specification and grade of fuel used in operation of the airplane is given in Servicing Diagram, figure 1–25.

An engine-driven fuel pump is provided on each engine. In addition, an electrically operated 28-volt dc booster pump is installed at the outlet of each wing fuel tank. The booster pumps aid the engine pumps in maintaining sufficient fuel pressure at the carburetor, and minimize the possibility of vapor lock. Should one engine become inoperative, the booster pumps are capable of delivering fuel to the other engine. Each engine fuel system has a vapor return line attached to the carburetor which returns vapor or unused fuel from the carburetor to the main fuel tanks. A check valve is installed in the vapor return line from the carburetor to prevent drainage of fuel from the main fuel tanks should a disconnection or severing of the line occur. On airplanes AF 53-7840 through 53-7884, AF 53-8153 through 53-8156, IK-450 through 466, and those on which T.O. 1C-119-538 has been accomplished, the vapor return lines from the carburetors are routed to the inboard fuel tanks. This prevents contamination of high grade fuel in the outboard tanks when alternate grade fuel is carried in the inboard tanks. On all other airplanes, the vapor return lines are connected to the outboard tanks. A fuel strainer is installed in the fuel supply line of each system between the firewall shut-off valve and the engine-driven fuel pump.



FUEL SYSTEM *Schematic*

Figure 1-7

To extend the operating range of this airplane for the purpose of remote tactical operations or extreme long range ferry missions, auxiliary fuel tanks may be installed. Two auxiliary fuel tanks are secured to the cargo compartment floor. The tanks are interconnected and are vented to the atmosphere through the side of the cargo compartment. A booster pump in the aft auxiliary tank is utilized to provide the required pressure for delivering fuel to the engines via the auxiliary fuel supply line which originates in the aft auxiliary tank and is connected to the crossflow line of the main system. Two manually operated valves are installed in this line to control the flow of auxiliary fuel; one at the aft end of the rear tank and the other in the cargo compartment ceiling, adjacent to the heater access. A check valve is installed between the two shut-off valves to prevent reverse flow of fuel from the main tanks.

FUEL SELECTOR SWITCHES.

Two four-position fuel selector wafer type switches (21, 32, figure 1–6), one for each engine fuel supply system, are installed on the overhead panel and electrically operate a 28-volt dc crossflow valve as well as individual fuel tank shut-off valves. The positions and functions of the fuel selector switches are shown in Figure 1–8.

BOOSTER PUMP SWITCHES.

Two fuel booster pump switches (2, figure 1–5), one left and one right, with NORMAL ON, OFF and EMERGENCY ON positions, are mounted on the pilot's switch panel. These pumps serve to supply fuel under pressure to the engine driven fuel pumps, the EMERGENCY ON position affording a greater fuel pressure than NORMAL ON. The center OFF position renders the booster pumps inoperative. Since the booster pump switches are wired through the fuel selector switches, they are operative only in the tank selected by the fuel selector switches.

FUEL, OIL AND HYDRAULIC SHUT-OFF SWITCHES.

A fuel, oil, and hydraulic shut-off switch (22, 31, figure 1–6) is provided for each engine. These switches are located on the overhead panel and remain in NORMAL position during normal operation with the

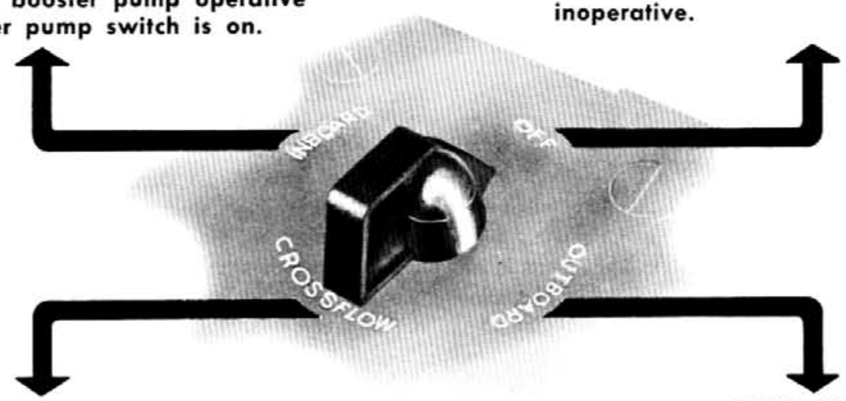
FUEL SELECTION

INBOARD . . .

Inboard tank valve and firewall shut-off valve . . . open. Outboard tank valve closed. Inboard booster pump operative whenever booster pump switch is on.

OFF . . .

Firewall shut-off valve and all tank shut-off valves . . . closed. Booster pumps are inoperative.



CROSSFLOW . . .

Inboard and outboard tank valves closed. Crossflow valve open, permitting fuel to flow from other side of system. Booster pumps are inoperative.

OUTBOARD . . .

Outboard tank valve and firewall shut-off valve . . . open. Inboard tank valve closed. Outboard booster pump operative whenever booster pump switch is on.

Note: *Positioning the Fuel, Oil and Hyd. Shut Off Switch to SHUT will shut off fuel flow at the firewall regardless of position of the Fuel Selector Switches.*

Figure 1-8

FUEL QUANTITY DATA

TANKS	NUMBER	USABLE FUEL	FULLY SERVICED	EXPANSION SPACE	TOTAL VOLUME
INBOARD	2	462 gal.	469 gal.	14 gal.	483 gal.
OUTBOARD	2	833 gal.	842 gal.	27 gal.	869 gal.

USABLE FUEL VS FLIGHT ATTITUDE

FLIGHT ATTITUDE	INBOARD TANKS	OUTBOARD TANKS	FLIGHT ATTITUDE	INBOARD TANKS	OUTBOARD
5° UP	460 gal.	827 gal.	5° DOWN	459 gal.	825 gal.
10° UP	453 gal.	815 gal.	10° DOWN	451 gal.	810 gal.
20° UP	440 gal.	796 gal.	20° DOWN	438 gal.	787 gal.
25° UP	427 gal.	768 gal.	25° DOWN	426 gal.	757 gal.

Figure 1-9

switch down and the switch guard closed. Should an emergency arise, they are placed in the SHUT position with the switch up and the switch guard open, causing all fuel, engine and hydraulic oil flow to be shut off at the firewall. The fuel selector switch in the OFF position also will shut off the fuel supply at the firewall.

AUXILIARY FUEL SHUT-OFF VALVE.

An auxiliary fuel system manually operated shut-off valve is located on the cargo compartment ceiling and is utilized when the auxiliary fuel tanks are installed in the cargo compartment. When the auxiliary fuel tanks are not installed, the valve is safety wired in the CLOSED position. When the auxiliary tanks are installed, this valve should be placed in the OPEN position at all times.

AUXILIARY FUEL TANK SHUT-OFF VALVE.

A manually operated shut-off valve is installed in the auxiliary fuel supply line at the aft end of the rear auxiliary tank. When using auxiliary fuel, the valve should be in the OPEN position permitting auxiliary fuel to enter the main fuel system. It should be placed in the CLOSED position after shifting back to the main fuel system to prevent air from being introduced into the main system through the auxiliary tanks after the tanks become empty.

AUXILIARY FUEL BOOSTER PUMP SWITCH.

An auxiliary fuel system booster pump switch is located on top of the main junction box on the right side of the cargo compartment. This switch is provided with ON and OFF positions and is turned ON when fuel is being used from the auxiliary fuel tanks. On airplanes with two main junction boxes, the auxiliary fuel booster pump switch is mounted on the left main junction box.

FUEL QUANTITY INDICATORS AND TEST BUTTONS.

Four electrically operated fuel quantity indicators (45, figure 1-22), are mounted on the instrument panel. These indicators are calibrated in pounds and register separate fuel level readings for each of the four main tanks. Capacitance-type tank units which function as the primary fuel quantity sensing devices are installed in the number 6 and 9 cells of the outboard tanks and in the number 2 cells of the inboard tanks. The 115-

volt ac tank units transmit a signal in proportion to the amount of fuel in the tank. This signal, after being amplified by the power units located on the auxiliary equipment floor, is directed to the indicators where it is represented in pounds of fuel available. If ac power fails, each fuel level indicator pointer will remain at its last reading, thus providing a reference for fuel calculation during the remainder of the flight. An indication is also given of fuel remaining in the tanks at the time the engines were shut down. Each indicator may be tested by a TEST button on the power unit housing. This causes the pointer to move counterclockwise if the indicator is functioning properly, and to remain still if electrical failure has occurred.

An additional test button is installed on the instrument panel immediately below each fuel quantity indicator. Depressing each of these buttons will give the same indication as obtained when the power unit button is depressed.

PUSH TO TEST

FUEL FLOWMETERS.

Two fuel flowmeters (47, figure 1–22), located on the instrument panel, are used to indicate fuel flow to the engines in pounds per hour (pph). A 26-volt ac fuel flow transmitter is installed in each engine fuel system between the engine driven fuel pump and the carburetor to electrically transmit the rate of flow to the indicator.

FUEL PRESSURE INDICATORS.

Two fuel pressure indicators (46, figure 1–22), one for the fuel supply of each engine, are located on the instrument panel. A fuel pressure transmitter is installed in each engine fuel system in the unmetered fuel chamber of the carburetor and transmits a signal, relative to the fuel pressure to which it is subjected, to the indicator where it is registered in pounds per square inch (psi).

DC ELECTRICAL POWER SYSTEM.

The airplane's 28-volt direct current electrical system is the single-wire, ground-return type generally employed in modern aircraft. A battery, two generators and an auxiliary power plant comprise the major power source components of the system. To assume the power load during ground operation, an external power receptacle is provided. The voltage in the system is automatically regulated to provide a constant output throughout the rated load range of the engine-driven generators. Heavy, interconnecting wiring between strategically located junction boxes forms the medium of dc power distribution.

Note

Several configurations of wiring are currently in effect on C-119G airplanes. For a specific airplane, these modifications should be borne in mind when consulting the text, electrical schematics, or emergency procedures.

The following airplanes have the emergency and monitor bus features:

AF—All

IK 458 thru 466

The following airplanes have emergency bus, but no monitor bus feature:

IK 433 thru 457

The following airplanes have neither emergency nor monitor bus features. They have no OVERRIDE position on the battery switch nor provision for automatic disconnection of the main bus from the battery in event of generator failure. Provisions, however, are incorporated for operation of jump lights, alarm bell, and spare instrument inverter directly from the battery while battery switch is OFF.

IK 441 and 442

The dc power distribution system is divided into several busses in such a manner that some non-essential equipment is automatically cut off in event of failure of one engine-driven generator. Failure of both engine-driven generators will automatically shut down all non-essential equipment unless the APP is operated to assume the load. Even with all generators out, essential equipment will operate directly from the battery or emergency bus. A second portion of the emergency bus supplying required flight instruments may be added to the battery load if necessary. Means are provided for operating any required equipment directly from the battery by employing special emergency procedures.

ELECTRICAL EQUIPMENT

Effective on airplanes:

AF 51-17365 thru AF 51-17367
AF 52-6000 thru AF 52-6003

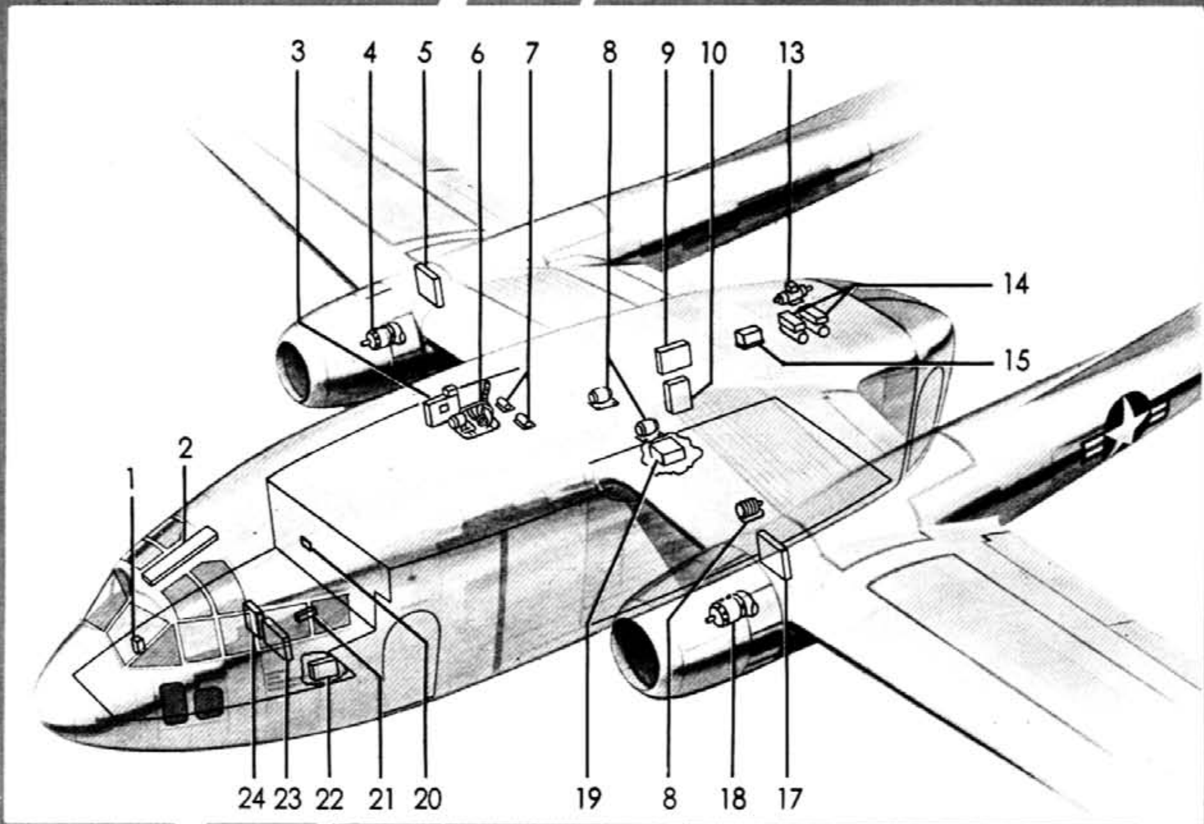
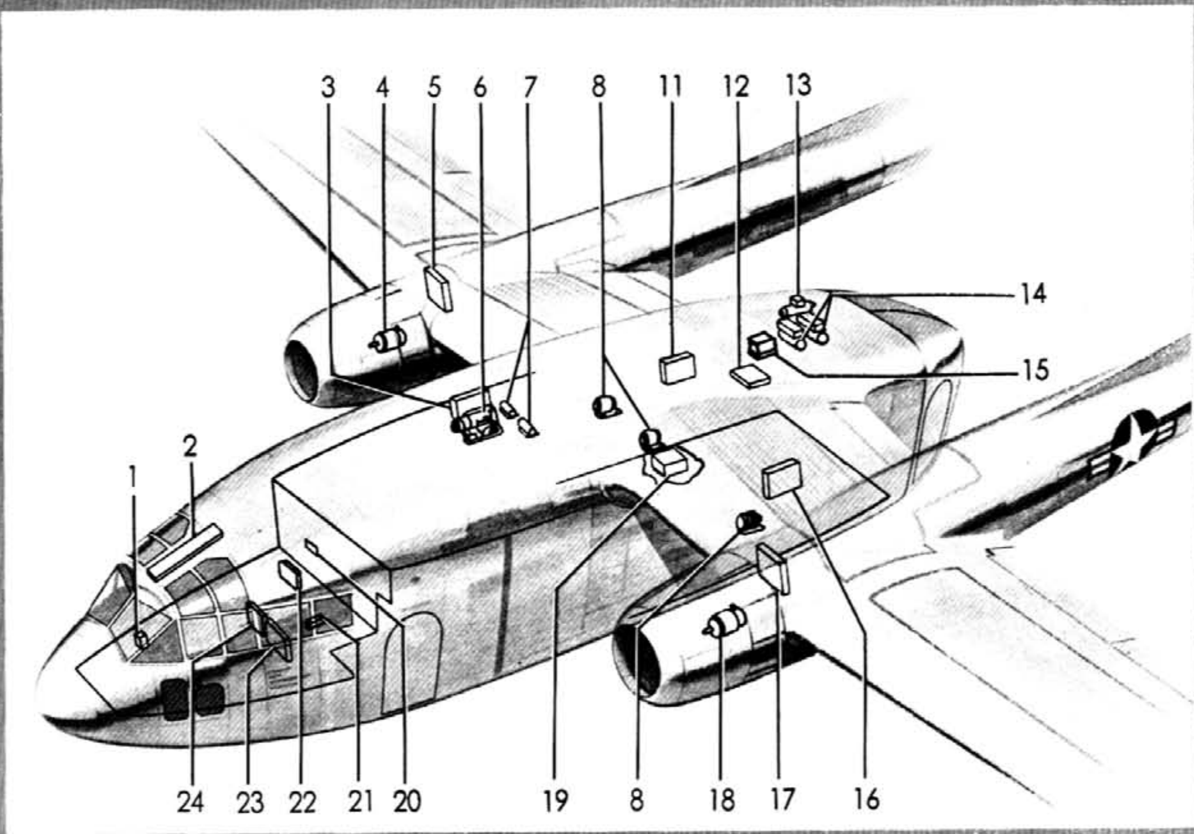


Figure 1–10



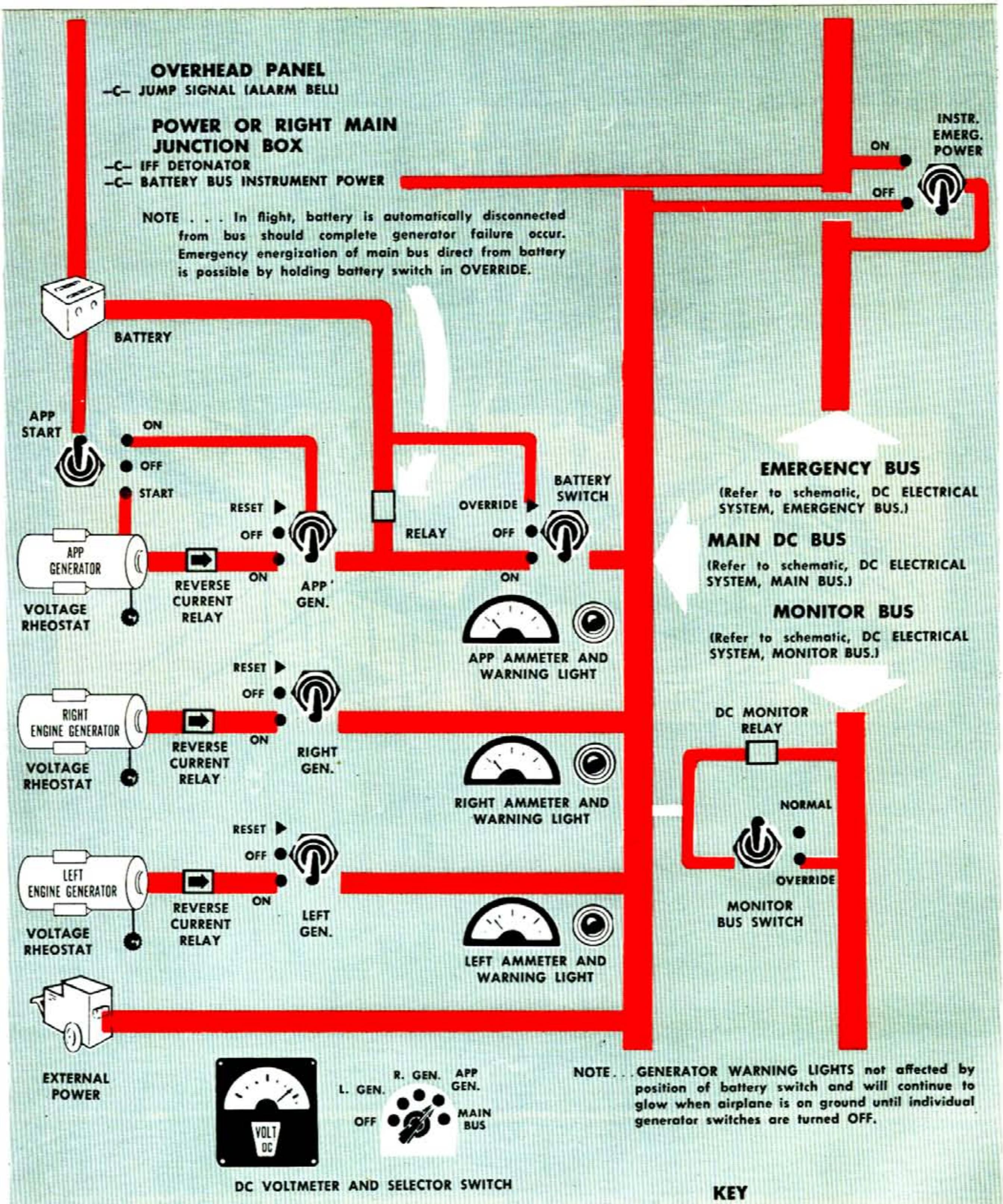
Effective on airplanes:

AF 51-8053 thru AF 51-8097
AF 52-5840 thru AF 52-5954
AF 52-6004 and subsequent.
IK 441 and subsequent.

1. EXTERNAL POWER RECEPTACLE
2. OVERHEAD PANEL
3. AUXILIARY FLOOR JUNCTION BOX
4. RIGHT ENGINE GENERATOR
5. RIGHT NACELLE JUNCTION BOX
6. AUXILIARY POWER PLANT
7. INSTRUMENT INVERTERS
8. VOLTAGE REGULATORS
9. POWER JUNCTION BOX
10. MAIN JUNCTION BOX
11. RIGHT MAIN JUNCTION BOX
12. OVERHEAD JUNCTION BOX
13. AUTOMATIC PILOT INVERTER
14. MAIN AND SPARE INVERTERS
15. AUTO PILOT JUNCTION BOX
16. LEFT MAIN JUNCTION BOX
17. LEFT NACELLE JUNCTION BOX
18. LEFT ENGINE GENERATOR
19. BATTERY
*20. MONITOR BUS SWITCH
*21. EMERG. CIRCUIT BREAKER PANEL
22. NOSE JUNCTION BOX
23. RADIO JUNCTION BOX
*24. MONITOR BUS JUNCTION BOX

* *Not on IK 441 and IK 442*

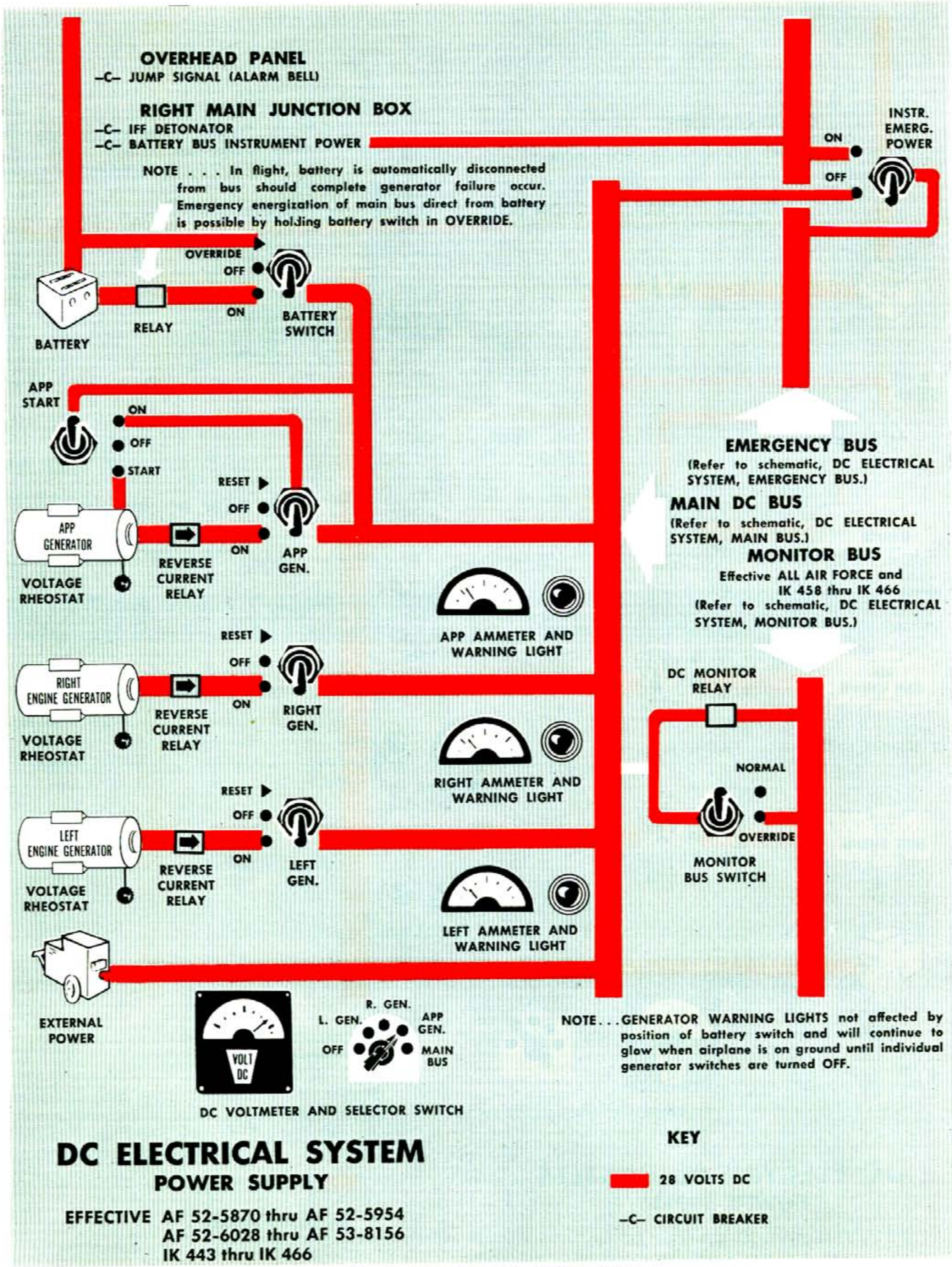
Figure 1–10



DC ELECTRICAL SYSTEM
POWER SUPPLY

EFFECTIVE AF 51-8053 thru AF 52-5869
AF 52-6000 thru AF 52-6027

Figure 1–11 (Sheet 1 of 3 Sheets)



DC ELECTRICAL SYSTEM

POWER SUPPLY

EFFECTIVE AF 52-5870 thru AF 52-5954
AF 52-6028 thru AF 53-8156
IK 443 thru IK 466

Figure 1–11 (Sheet 2 of 3 Sheets)

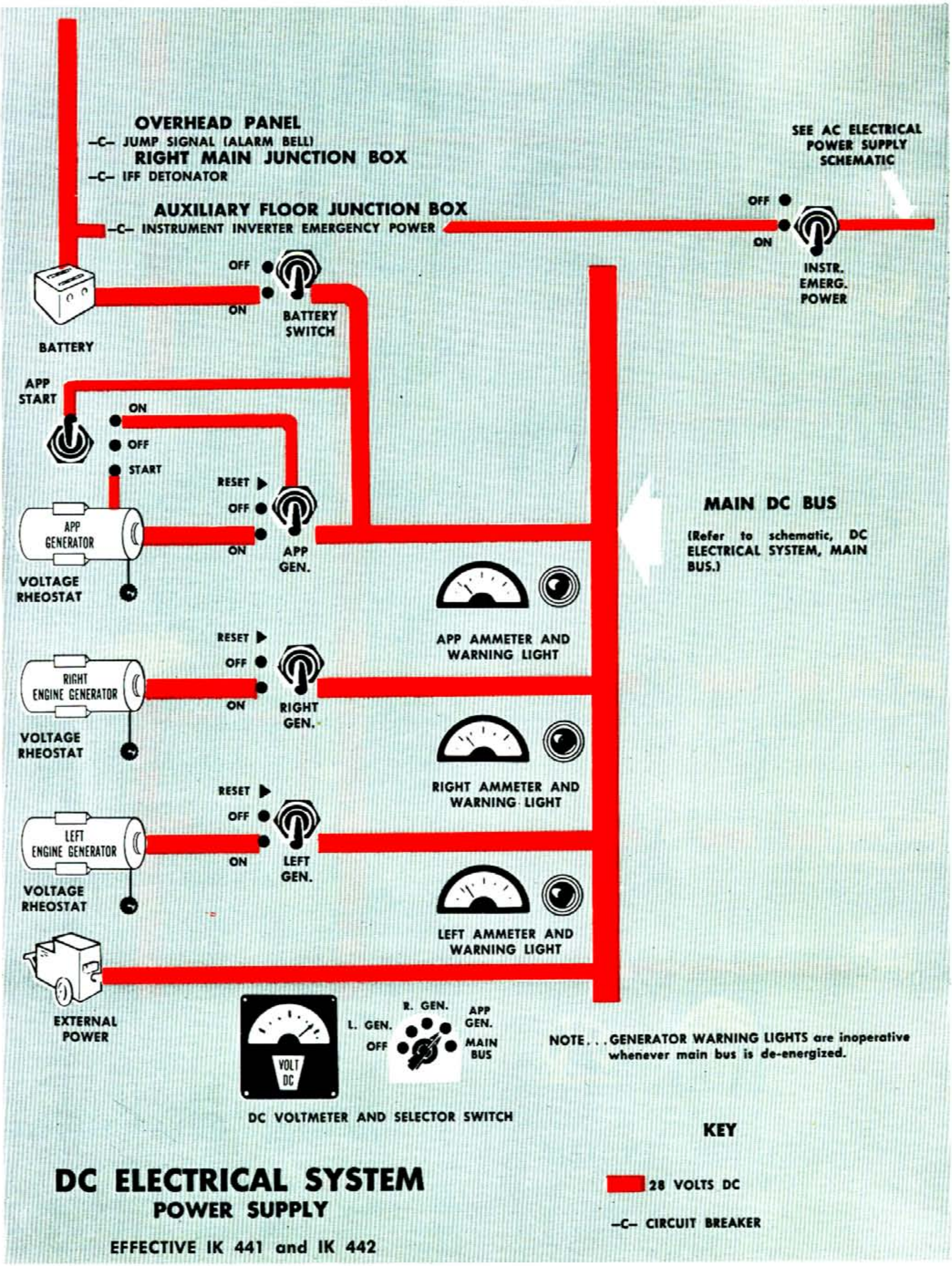


Figure 1–11 (Sheet 3 of 3 Sheets)

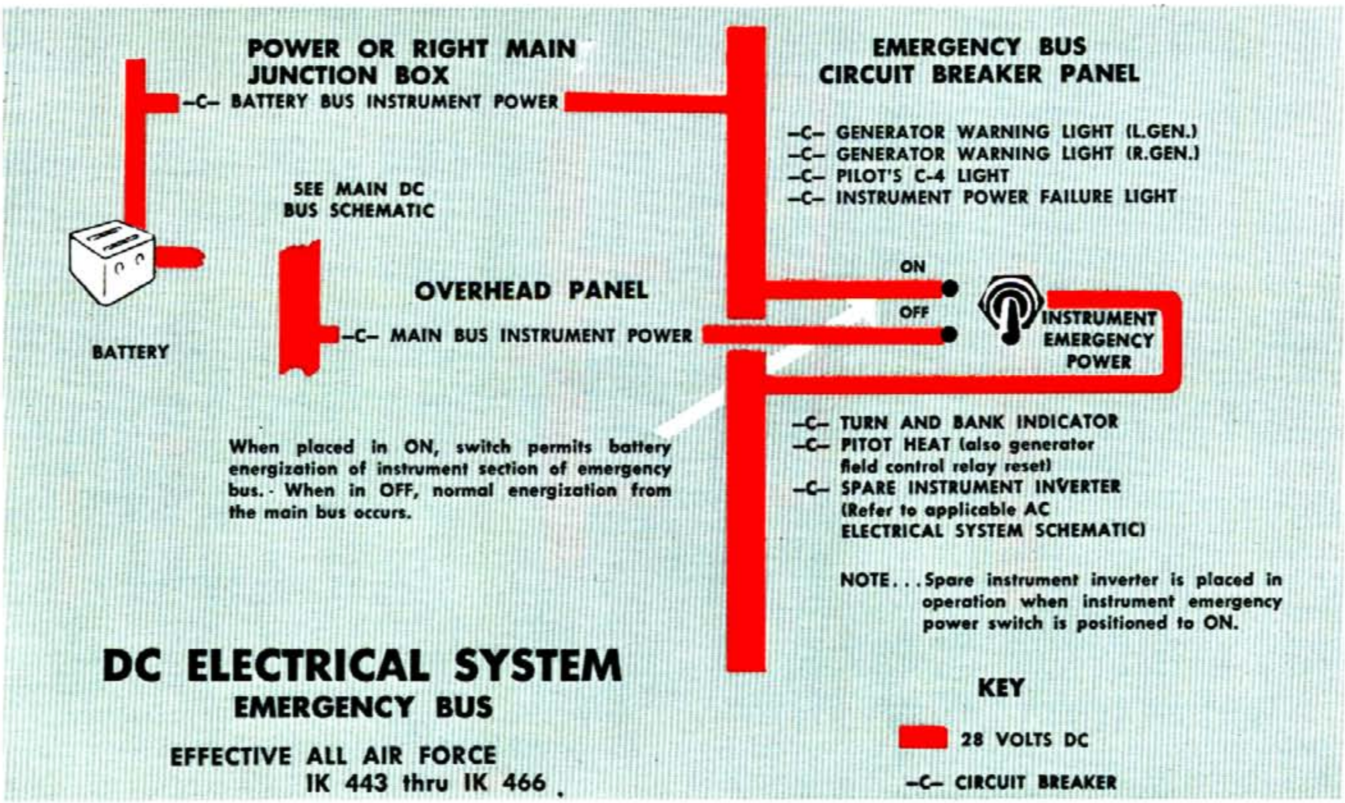


Figure 1–11A

FLIGHT EMERGENCY BUS.
(On all airplanes except IK 441 and 442).

A flight emergency bus is installed to conserve battery power for the energization of essential flight equipment under emergency conditions. Should all generators fail in flight, the battery output is automatically disconnected from the main dc bus; only the following units on the flight emergency bus remain operative: the instrument power failure warning light, both engine generator power failure lights, and the pilot's C-4 spotlight. Power for operation of these units is supplied directly from the battery regardless of the setting of the battery switch. Additional equipment on the flight emergency bus may be energized by battery power when the emergency power switch is turned to the ON position. This equipment includes pilot's turn-and-bank gyro, spare instrument inverter (power source of pilot's attitude and directional gyros) and pitot heaters; power is also made available for resetting the engine generator field control relays. The flight emergency bus and its associated circuit breakers are located at the flight mechanic's station. A ground cut-off relay, operated by the left main landing gear oleo switch, is incorporated into the system to prevent a drain of the battery when the airplane is in a parked power-off condition. With the airplane on the ground and the oleo strut compressed, the ground cut-off relay is tripped, turning off the instrument power failure light and the pilot's C-4 spotlight. These units would otherwise remain energized and discharge the battery. The engine generator failure lights will contine to glow until the generator control switches are placed in the OFF position.

DC MONITOR BUS.
(On all airplanes except IK 441 thru 457).

To prevent overloading an engine-driven generator during operation with only one generator operative, provisions have been incorporated on some airplanes to disconnect automatically certain radio, radar, and propeller de-icing equipment from the main dc bus. The units thus affected by failure of an engine-driven generator are the liaison transmitter-receiver, Loran, APN-12, the APS-42, when installed, and, on airplanes AF 53-7880 thru 53-7884, propeller de-icing. An ac monitor bus is similarly disconnected at the same time and ac power to the APN-12 and Loran sets is interrupted. A monitor bus switch for energizing the above equipment under partial power conditions is provided on the radio rack.

BATTERY.

A 24-volt, 36 ampere-hour aircraft storage battery is installed under the cargo compartment floor at the lower right side of the fuselage in approximately the same longitudinal location as the wing flap leading edge. The battery is accessible from the outside of the airplane through a placarded hinged panel. Emer-

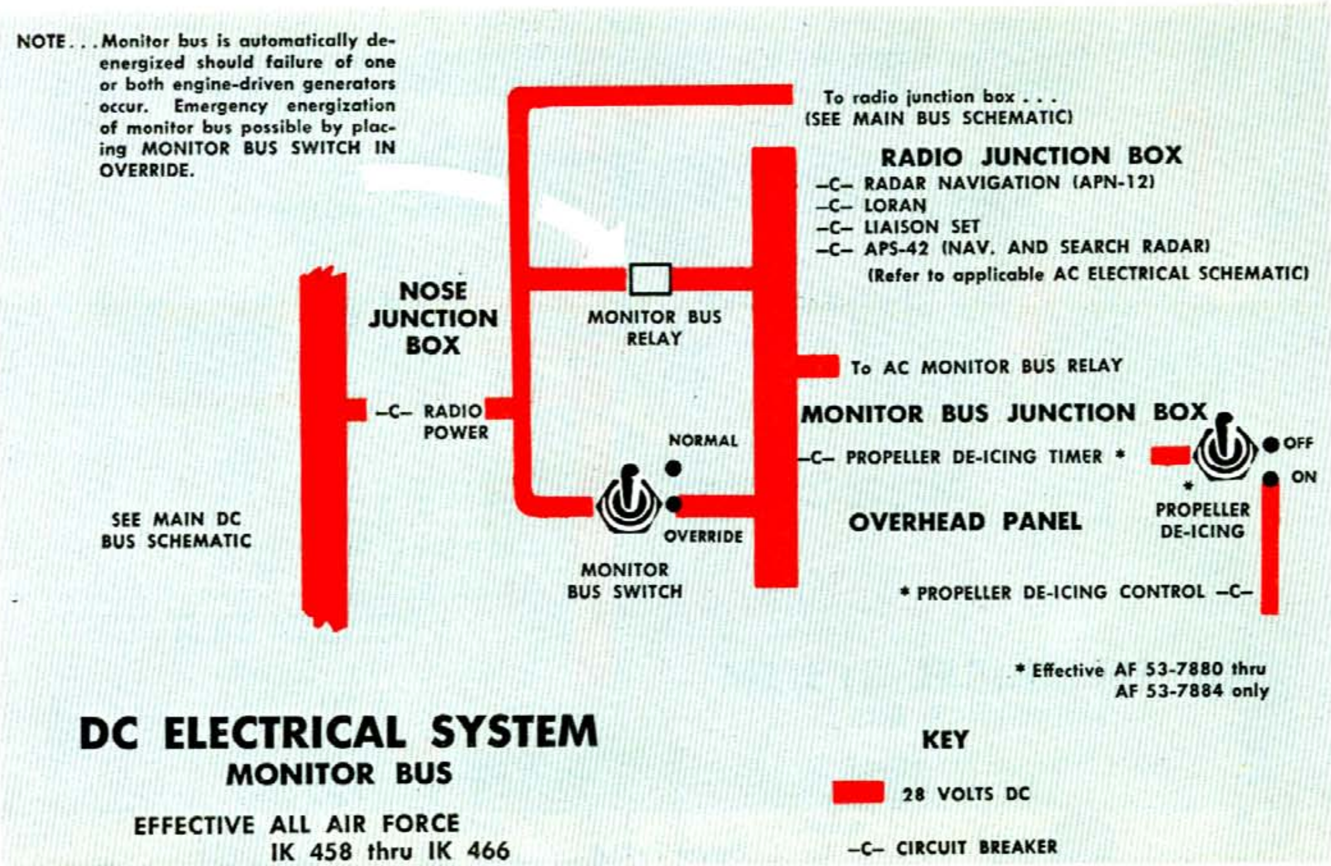


Figure 1–11B

gency quick-disconnection of the battery is possible from inside the cargo compartment after removal of a rectangular access door behind the troop seats near the right main junction box. A forced vent system drives battery fumes through a pint jar battery sump where the fumes are neutralized. The battery is used for ground starting of the APP when no external power is available, and to provide an emergency source of power at all times if the generators should fail. Emergency provisions are made for operation of the generator warning lights, instrument power failure light, pilot's C-4 light, IFF destructor, alarm bell, and jump lights directly from the battery. Flight instruments, pitot heat, and generator field control relay reset may also be operated directly from the battery, or the entire main bus may be temporarily energized by the battery through employment of the appropriate emergency procedures.

ENGINE-DRIVEN GENERATORS.

A 300-ampere, engine-driven, wide-speed range, direct current generator is mounted on the accessory drive section of each engine. These generators meet the airplane's total electrical power demands during normal flight operation by supplying the dc bus, driving the ac inverters, and recharging the battery. A constant 28-volt dc output is delivered from the generators at engine speeds of 1200 rpm and above.

AUXILIARY POWER PLANT.

An auxiliary power plant located in the auxiliary equipment compartment consists of a 28-volt, 175 ampere generator driven by an internal combustion engine. This unit may be electrically or manually started when dc power demands require its operation. The APP is ordinarily operated during take-offs and landings under night and/or IFR conditions or during emergency operations. If no external power source is available, the APP may be used to start the engines and supply power for ground checks. For description of APP controls and operation, refer to Section IV.

Note

While the generator is rated at 200 amperes, only 175 amperes are continuously available because of the horsepower limitations of the driving engine.

EXTERNAL POWER RECEPTACLE.

An external power receptacle on the right side of the fuselage below the crew compartment side windows permits the use of an external generator to supply power to the dc bus for all ground operation.

CIRCUIT BREAKER PANELS.

A circuit breaker panel (Figure 1-13) at the aft end of the overhead panel contains circuit breakers for most of the electrical control circuits in the airplane. Emergency bus circuit breakers are installed on the emergency bus circuit breaker panel (Figure 1-13) located just forward of the APP controls. Each circuit breaker is suitably placarded to indicate its function in the electrical system. Circuit breakers are also installed in the main, power, nose, auxiliary floor, nacelle, overhead, monitor bus, and radio junction boxes in such a manner that they are accessible from the exterior of the box. These circuit breakers, likewise, are individually identified. The circuit breakers in the nacelle junction boxes are not accessible in flight. Refer to the DC Electrical System Diagram (Figure 1-11) for a complete listing and location of the circuit breakers.

Note

Circuit breakers are installed to protect the various components of the dc electrical system and should not be utilized as switches.

BATTERY SWITCH. (On all airplanes except the IK 441 and 442).

A three-position, ON-OFF-OVERRIDE, battery switch (10, figure 1-6) on the overhead panel controls a relay connecting the battery to the main dc bus. In the OFF position, the battery is disconnected from the main dc bus. In the ON position, the battery is connected to the main dc bus if the airplane is on the ground, or if any of the airplane's generators are on the bus. In the OVERRIDE position, the battery is connected to the main dc bus regardless of the operation of the generators. The ON position is normally used to permit the generator-supplied bus to charge the battery or for ground operation of the APP when no external power is available. The momentary-contact OVERRIDE position is an emergency position permitting the battery to supply the main bus.

CAUTION

To prevent rapid depletion of the battery all unnecessary equipment should be individually turned off before the OVERRIDE position is used. The OVERRIDE position should not be used for starting the APP.

BATTERY SWITCH. (On airplanes IK 441 and 442).

A two-position, ON-OFF, battery switch on the overhead panel connects or disconnects the battery and the main dc bus. In the ON position the battery is connected to the bus. This maintains the battery in a charged condition when generators are operative and supplies power for starting the APP on the ground. The OFF position disconnects the battery from the main dc bus.

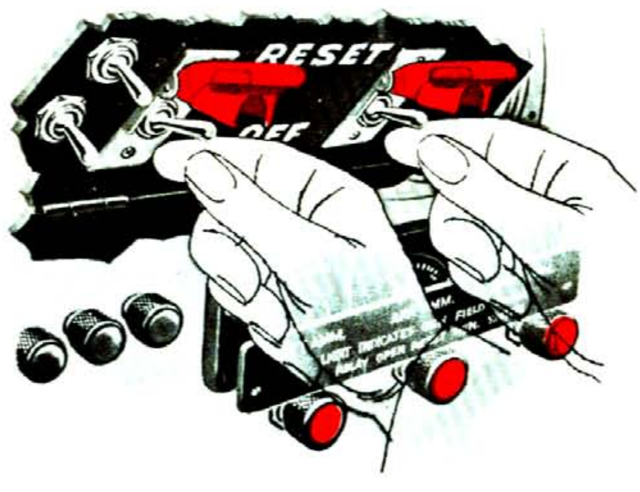
EMERGENCY POWER SWITCH. (On all airplanes except IK 441 and 442).

An emergency power switch (27, figure 1-22) on the instrument panel is a two-position switch with OFF and ON positions. If the switch is turned to the ON position when all generator power has been lost, battery power is made available to the flight instruments section of the emergency bus for operation of the following components: pilot's turn-and-bank gyro, pitot heaters, generator field control relay reset, and spare instrument inverter (power source of pilot's attitude and directional gyros). When the switch is in the OFF position and the main dc bus is energized as in normal operation, power for energizing the above equipment is obtained from the main dc bus.

EMERGENCY POWER SWITCH. (On airplanes IK 441 and 442).

An emergency power switch (27, figure 1-22) on the instrument panel permits the pilot's attitude and directional gyros to be used if the airplane's electrical generators should fail. By turning the switch ON, dc power is obtained from the battery to energize the spare instrument inverter, thus providing power for operation of the pilot's attitude and directional gyros. The battery switch need not be ON to obtain battery voltage to energize the spare instrument inverter if the emergency power switch is used; power is obtained directly from the battery.

GENERATOR SWITCHES.



Left and right engine generator switches (33, figure 1-6) on the overhead panel control the 28-volt, direct current generator mounted on each engine. The posi-

OVERHEAD PANEL

CARGO HEAT MASTER SWITCH

OFF
ON

- –C– CABIN BLOWERS (CONTROL)
- –C– HEATER CONTROL (PRIMARY FUEL)
- –C– HEATER VALVES (HEATER OPERATION)

- –C– ANTI-ICING (INDICATORS)
- –C– AUTOMATIC PILOT (INVERTER)
- –C– AUXILIARY INSTRUMENT LIGHTS
- –C– BANK AND TURN (INDICATOR)
- –C– CARBURETOR AIR CONTROL
- –C– CARBURETOR AIR TEMPERATURE
- –C– CARGO HEAT
- –C– COCKPIT AIR
- –C– CONTROL LAMP (ALDIS)
- –C– COWL FLAPS
- –C– CROSSFLOW (VALVES)
- –C– CYLINDER HEAD TEMPERATURE
- –C– DOME LIGHTS
- –C– DOOR WARNING LIGHTS
- –C– ELEVATOR LOCK—NOSE STEERING
- –C– ENGINE ACCESSORY HEAT
- –C– FIRE DETECTOR
- –C– FIRE EXTINGUISHER
- –C– FORMATION LIGHTS
- –C– FUEL AND OIL SHUT-OFF (VALVES)
- –C– FUEL BOOSTER PUMPS (CONTROL)
- –C– FUEL PRIMER—OIL DILUTION
- –C– INSTRUMENT INVERTER
- –C– INSTRUMENT LIGHTS
- –C– LANDING GEAR CONTROL
- –C– LANDING GEAR INDICATION (LIGHTS)
- –C– LANDING GEAR WARNING (HORN)
- –C– LANDING LIGHTS
- –C– MAIN BUS INSTRUMENT POWER
- –C– MAIN INVERTER
- –C– MONORAIL SALVO
- –C– NAVIGATION LIGHTS
- –C– OIL COOLER CONTROL
- –C– OIL DILUTION—FUEL PRIMER
- –C– OIL LEVEL
- –C– OIL TEMPERATURE
- –C– PASSING LIGHT
- –C– PEDESTAL FRICTION

- –C– PITOT HEAT
- –C– PROPELLER DE-ICING TIMER

PROPELLER DE-ICING

OFF
ON

- –C– PROPELLER DE-ICING

- –C– RADOME ANTI-ICING
- –C– STARTER
- –C– SUPERCHARGER
- –C– TEST POWER INDICATOR LIGHTS
- –C– THROTTLE STOPS
- –C– TRIM TAB CONTROL
- –C– TRIM TAB (INDICATORS)
- –C– TURN & BANK INDICATOR
- –C– WATER INJECTION
- –C– WINDSHIELD ANTI-ICING
- –C– WINDSHIELD WIPERS
- –C– WING AND TAIL ANTI-ICING
- –C– WING FLAPS

POWER JUNCTION BOX Ⓐ

- –C– AUTO PILOT INVERTER
- –C– AUTO PILOT POWER
- –C– FUEL BOOSTER PUMPS (POWER)
- –C– INVERTER FAILURE (AUTO PILOT)
- –C– MAIN INVERTER (POWER)
- –C– MONORAIL BRAKE
- –C– MONORAIL SALVO
- –C– SPARE INVERTER (POWER)
- –C– SPARE INVERTER CONTROL (CHANGEOVER)
- –C– VOLTMETER (MAIN BUS)

MAIN JUNCTION BOX Ⓐ

- –C– ACCESSORY COMPARTMENT DOME LIGHT
- –C– CABIN BLOWERS (POWER)
- –C– CARGO DOME LIGHTS
- –C– FLOOR LIGHTS
- –C– HEATER FUEL VALVES
- –C– NAVIGATOR'S TABLE LIGHT
- –C– WATER INJECTION CONTROL
- –C– WATER INJECTION PUMP

LEFT JUNCTION BOX Ⓑ

- –C– CABIN BLOWER (POWER)
- –C– FUEL BOOSTER PUMPS (POWER)
- –C– HEATER FUEL VALVE
- –C– WATER INJECTION PUMP

FOR ORIGIN OF THIS CIRCUIT REFER TO APPLICABLE DC POWER SUPPLY SCHEMATIC.

Ⓐ Effective AF 51-17365 thru AF 51-17367
AF 52-6000 thru 52-6003

DC ELECTRICAL SYSTEM
MAIN BUS
EFFECTIVE ALL AIRPLANES

KEY

(red) 28 VOLTS DC

–C– CIRCUIT BREAKER

Figure 1–12

RIGHT MAIN JUNCTION BOX (B)

–C– APX-6 TEST
–C– CABIN BLOWER (POWER)
–C– FLOOR LIGHTS
–C– FUEL BOOSTER PUMPS (POWER)
–C– HEATER FUEL VALVE
–C– TEST RECEPTACLE APX-25
–C– WATER INJECTION PUMP
–C– WATER QUANTITY GAGE

OVERHEAD JUNCTION BOX (B)

–C– AUTO PILOT INVERTER
–C– AUTO PILOT POWER
–C– INVERTER FAILURE
–C– MAIN INVERTER (POWER)
–C– MONORAIL BRAKE
–C– MONORAIL SALVO
–C– PROPELLER REVERSING STOP
–C– SPARE INVERTER (POWER)
–C– SPARE INVERTER CONTROL (CHANGEOVER)

NOSE JUNCTION BOX

–C– ACCESSORY COMPARTMENT DOME LIGHTS
–C– CARGO DOME LIGHTS
–C– FLIGHT OPERABLE DOOR
–C– NAVIGATOR'S TABLE LIGHT
–C– RADIO POWER (B)
–C– VOLTMETER

AUXILIARY FLOOR JUNCTION BOX

–C– AMMETER (APP)
–C– GENERATOR FIELD (APP)
–C– GENERATOR RELAY (APP)
–C– GENERATOR WARNING LIGHT (APP)
–C– IGNITION ANALYZER
–C– INSTRUMENT INVERTER LIGHT (BOTH)
–C– MAIN INSTRUMENT INVERTER (POWER)
–C– RADIO POWER (A)
–C– SPARE INSTRUMENT INVERTER (POWER)
–C– VOLTMETER (APP)

RIGHT NACELLE JUNCTION BOX

–C– AMMETER (R. GEN)
–C– COWL FLAP MOTOR
–C– DE-ICING AMMETER (PROPELLER)
–C– GENERATOR (CONTROL)
–C– GENERATOR WARNING LIGHT (R. GEN)
–C– LANDING LIGHT
–C– NACELLE PREHEAT
–C– VOLTMETER (R. GEN)

LEFT NACELLE JUNCTION BOX

–C– AMMETER (L. GEN)
–C– COWL FLAP MOTOR
–C– DE-ICING AMMETER (PROPELLER)
–C– GENERATOR (CONTROL)
–C– GENERATOR WARNING LIGHT (L. GEN)
–C– LANDING LIGHT
–C– NACELLE PREHEAT
–C– VOLTMETER (L. GEN)

MONITOR BUS JUNCTION BOX (C)

–C– PROPELLER DE-ICING TIMER

–C– MONITOR BUS CONTROL

RADIO JUNCTION BOX

–C– ALTIMETER
–C– APS-42 PRESSURE PUMP
–C– APS-42 RADAR
–C– AUTO COMPASS
–C– AUTO COMPASS NO. 1
–C– AUTO COMPASS NO. 2
–C– EMERGENCY KEYER
–C– GLIDE PATH RECEIVER
–C– HOMING ADAPTER
–C– IFF TRANSMITTER
–C– INTERPHONE
–C– LIAISON DYNAMOTOR
–C– LIAISON RECEIVER
–C– LORAN
–C– MARKER RECEIVER
–C– RADAR NAVIGATION
–C– UHF COMMAND
–C– VHF COMMAND
–C– VHF NAVIGATION
–C– VHF NAVIGATION TEST

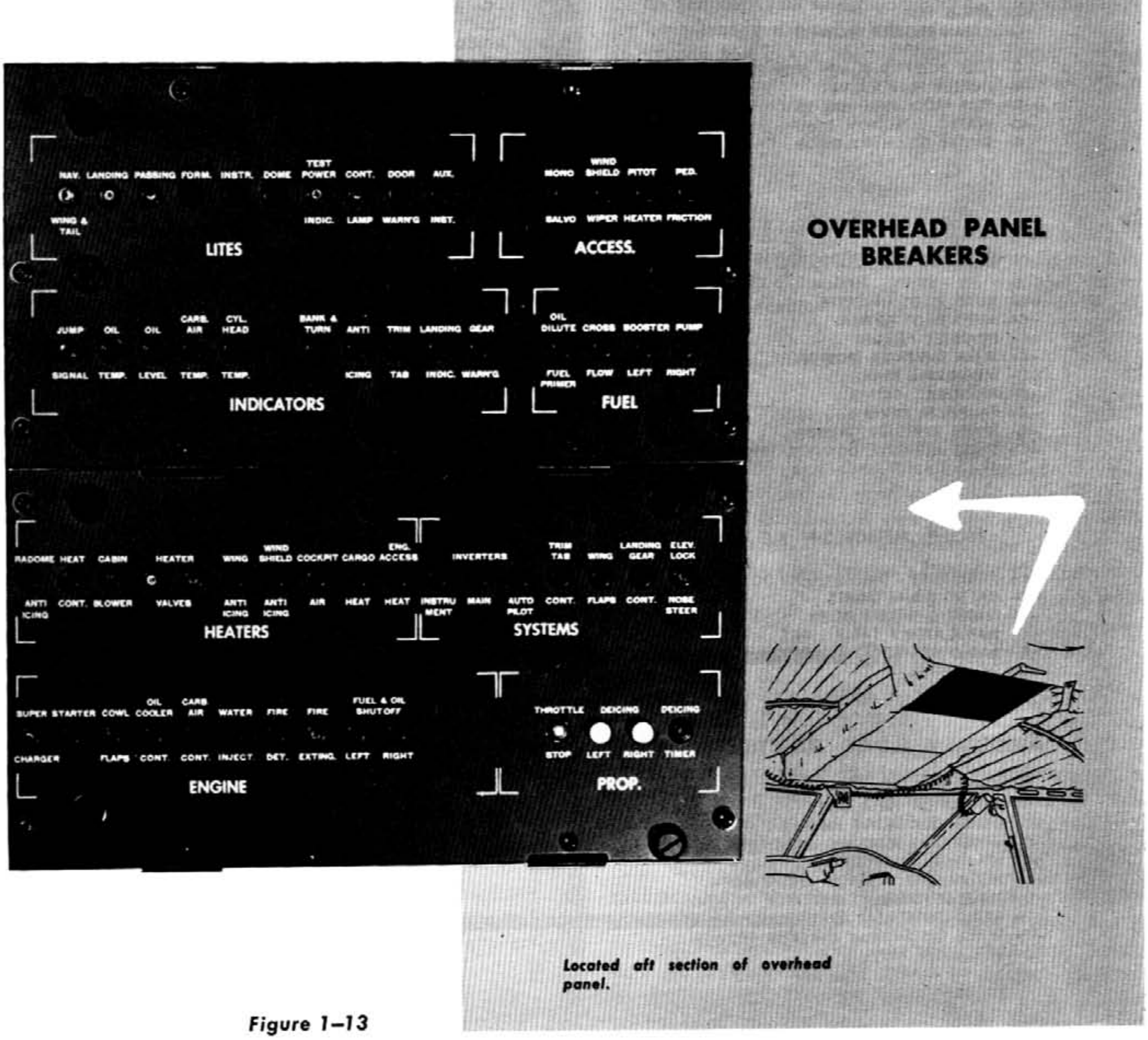
(B) Effective AF 51-8053 thru AF 51-8097
AF 52-5840 thru AF 52-5954
AF 52-6004 thru AF 53-8156
IK 441 thru 466

(C) Effective ALL AIR FORCE and
IK 458 thru IK 466

NOTE... More circuit breakers may be found listed here than occur in a specific panel or junction box. This listing represents a complete composite of all C-119G airplanes. On any specific airplane, the circuit breakers are clearly placarded at the junction box or panel in which they are located.

Figure 1–12

CIRCUIT BREAKER PANELS



Located aft section of overhead panel.

Figure 1–13

tions and corresponding functions of the generator control switches are as follows:

OFF—The output of the generator is excluded from system.

ON—The output of the generator is permitted to energize the system.

RESET—Permits the generator output to energize the system after the current flow has been interrupted because of an excessive voltage condition which tripped the generator field control relay.

Note

Should all generators fail on emergency bus airplanes, voltage for resetting the field control relays is not available until the instrument emergency power switch is placed in the ON position. The generator reset circuit utilizes the same circuit breaker as does the pitot heat circuit. Therefore, the pitot heat circuit breaker must be closed when resetting engine generator field control relays.

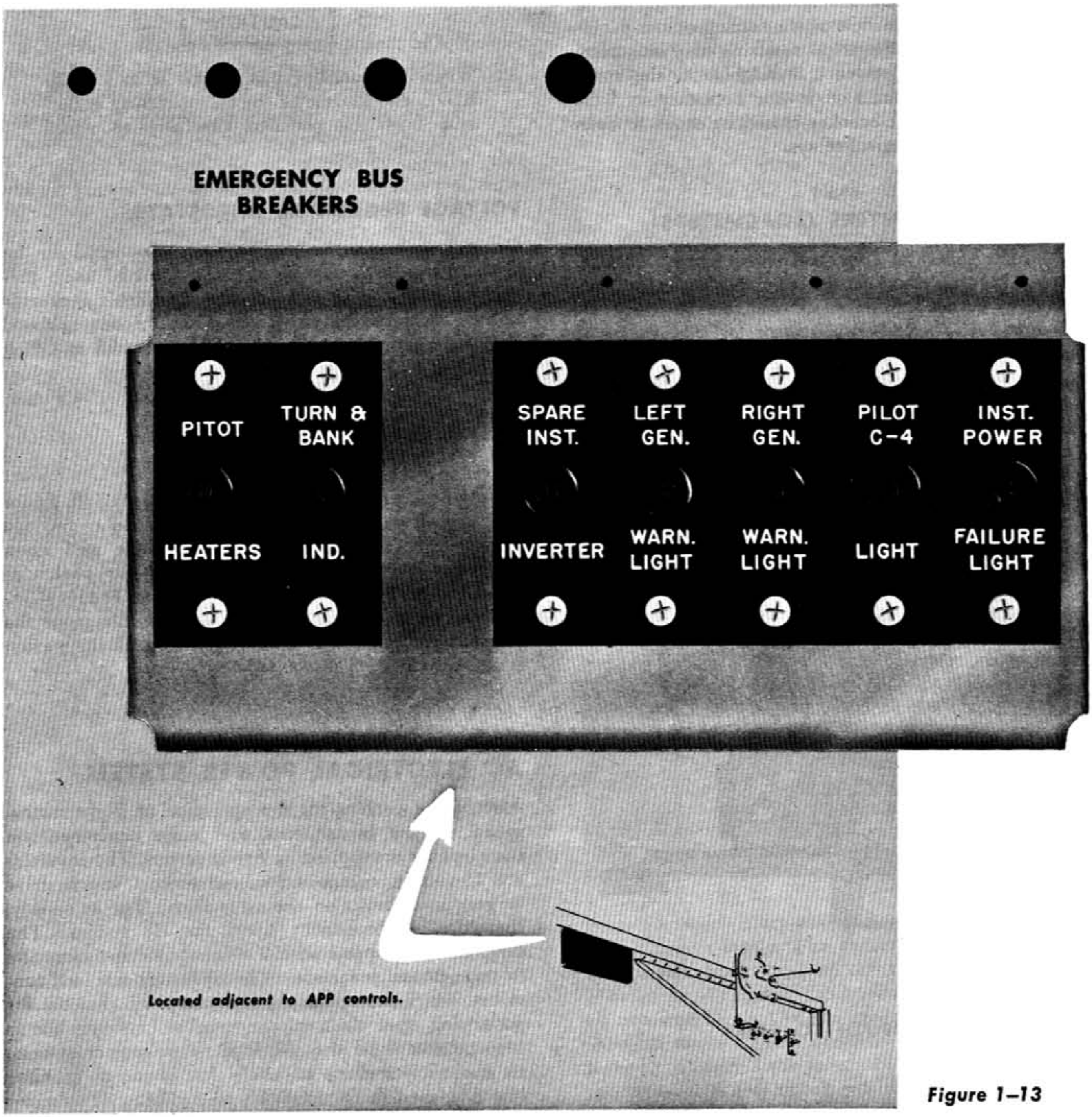


Figure 1–13

ENGINE GENERATOR WARNING LIGHTS. (On all airplanes except IK 441 and 442).

A warning light (32, 34, figure 1-22) for each engine generator is located on the instrument panel. The illumination of a generator power failure warning light indicates an excessive voltage condition, under-voltage, or a mechanical failure of the generator. The lights are operative only when the respective generator has been disconnected from the bus and its generator control switch is in the ON position. The generator warning lights are not affected by the position of the battery switch.

Note

Generator power failure lights also indicate that the generators have not cut in; this condition may occur when taxiing at a low rpm.

ENGINE GENERATOR WARNING LIGHTS. (On airplanes IK 441 and 442).

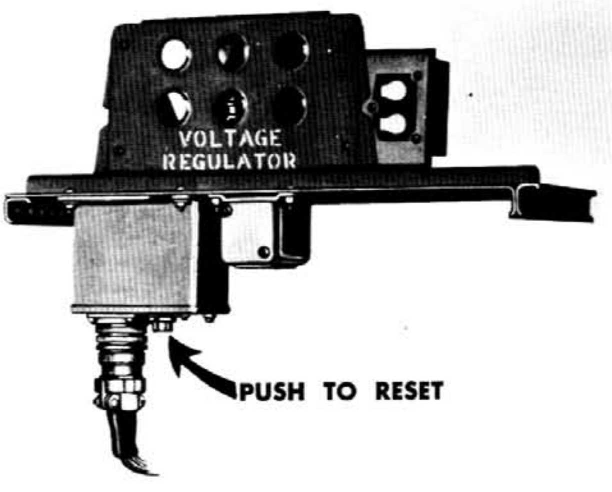
The engine generator warning lights indicate only an over-voltage condition which has tripped the corresponding generator field control relay. Warning lights on these airplanes are placarded "Light Indicates Gen. Field Control Relay Open." The indication will be obtained regardless of the setting of the generator control switch. Electrical resetting may be accomplished as long as power is available on the main bus. On these airplanes it is not necessary to have the pitot heat circuit breaker closed in order to reset the generator field control relays.

GENERATOR AMMETERS (LOADMETERS).

Each of the generator ammeters (loadmeters), (32, 34, figure 1-22), on the instrument panel measures current flow but is calibrated to indicate the load assumed by its respective generator in percentage of the generator's rated capacity.

GENERATOR FIELD CONTROL RELAY MANUAL RESET BUTTONS.

A reset button is incorporated into each of the field control relays. The right engine field control relay is located on the right side of the fuselage; the left engine field control relay, on the left side. Should a field control relay fail to reset when its corresponding generator switch is placed in the RESET position, the relay may be manually reset by depressing the reset button on the relay.



MONITOR BUS SWITCH.
(On all airplanes except IK 441 thru 457).

A two-position switch on the navigator's utility shelf with NORMAL and OVERRIDE positions serves to re-energize the ac and dc monitor busses after they have been de-energized due to the loss of one engine-driven generator. In NORMAL position, the monitor busses are energized as long as both engine-driven generators are operative; de-energized when either or both generators fail. In OVERRIDE position, the monitor busses are energized whenever the main ac and dc busses are energized.

CAUTION

When the monitor bus switch is in OVERRIDE, the loadmeter for the operating generator must be watched carefully to avoid overloading.

VOLTAGE REGULATOR RHEOSTATS.

On each generator voltage regulator mounted on the sides of the cargo compartment, a knurled knob permits operation of a rheostat which controls generator voltage output. These rheostats should be regulated only by qualified personnel except as stated in GENERATOR FAILURE, Section III.

DC VOLTMETERS AND DC VOLTAGE SELECTOR SWITCH.

A dc voltmeter and voltage selector switch (30, figure 1-22) are installed on the instrument panel to provide engine generator, auxiliary power plant, and main bus voltage readings. When the selector switch is turned to APP GEN, R. GEN, L. GEN, or MAIN BUS, the voltmeter will indicate the output voltage of the selected component. An OFF position of the switch is also provided.

AC ELECTRICAL POWER SYSTEM.

Alternating current for the operation of flight instruments, engine instruments, and radio equipment on the airplane is supplied by five inverters. The inverters are 28-volt dc compound-wound motors which drive ac generators built on the same shaft. The ac supply, therefore, is dependent upon the dc system. The airplane is equipped with 2 single-phase inverters and 3 three-phase inverters. The inverters are self-contained units incorporating the necessary controls for governing rpm and voltage. An ac monitor bus is incorporated into the single-phase ac power system; its operation and use parallels that of the dc monitor bus described in DC POWER SYSTEM.

SINGLE-PHASE INVERTERS.

Two single-phase, 1500 volt-ampere, 115-volt, 400 cycle inverters, designated main and spare, are located in the aft end of the cargo compartment on a shelf above the soundproofing and supply ac power required for operation of autosyn instruments, magnetic compass light, ignition analyzer, and certain communications equipment. In normal operation the main inverter is used; should it become inoperative, an automatic changeover unit switches the spare inverter into the system. On airplanes equipped with AN/APS-42 equipment, the spare single-phase in-

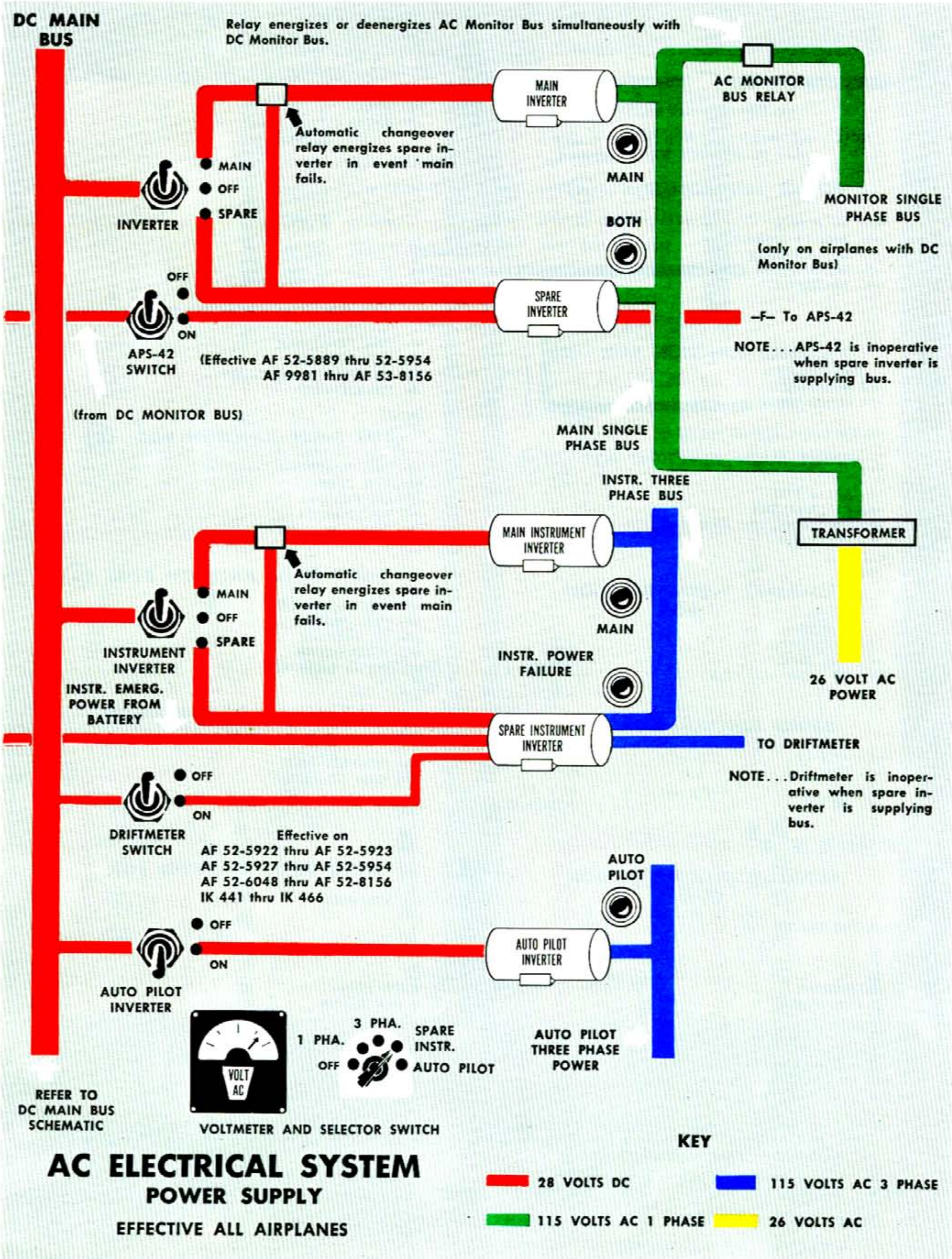

DC MAIN BUS
Relay energizes or deenergizes AC Monitor Bus simultaneously with DC Monitor Bus.
MAIN INVERTER
AC MONITOR BUS RELAY
Automatic changeover relay energizes spare inverter in event main fails.
MAIN
OFF
SPARE
INVERTER
MAIN
BOTH
MONITOR SINGLE PHASE BUS
(only on airplanes with DC Monitor Bus)
OFF
ON
SPARE INVERTER
—F— To APS-42
APS-42 SWITCH
(Effective AF 52-5889 thru 52-5954 AF 9981 thru AF 53-8156
NOTE . . . APS-42 is inoperative when spare inverter is supplying bus.
(from DC MONITOR BUS)
MAIN SINGLE PHASE BUS
INSTR. THREE PHASE BUS
TRANSFORMER
MAIN INSTRUMENT INVERTER
Automatic changeover relay energizes spare inverter in event main fails.
MAIN
OFF
SPARE
MAIN
INSTRUMENT INVERTER
INSTR. POWER FAILURE
26 VOLT AC POWER
INSTR. EMERG. POWER FROM BATTERY
SPARE INSTRUMENT INVERTER
TO DRIFTMETER
OFF
ON
NOTE . . . Driftmeter is inoperative when spare inverter is supplying bus.
DRIFTMETER SWITCH
Effective on
AF 52-5922 thru AF 52-5923
AF 52-5927 thru AF 52-5954
AF 52-6048 thru AF 52-8156
IK 441 thru IK 466
AUTO PILOT
OFF
ON
AUTO PILOT INVERTER
AUTO PILOT INVERTER
3 PHA.
1 PHA.
SPARE INSTR.
OFF
AUTO PILOT
VOLT AC
AUTO PILOT THREE PHASE POWER
REFER TO DC MAIN BUS SCHEMATIC
VOLTMETER AND SELECTOR SWITCH
KEY
AC ELECTRICAL SYSTEM
POWER SUPPLY
EFFECTIVE ALL AIRPLANES
28 VOLTS DC
115 VOLTS AC 3 PHASE
115 VOLTS AC 1 PHASE
26 VOLTS AC

Figure 1–13A

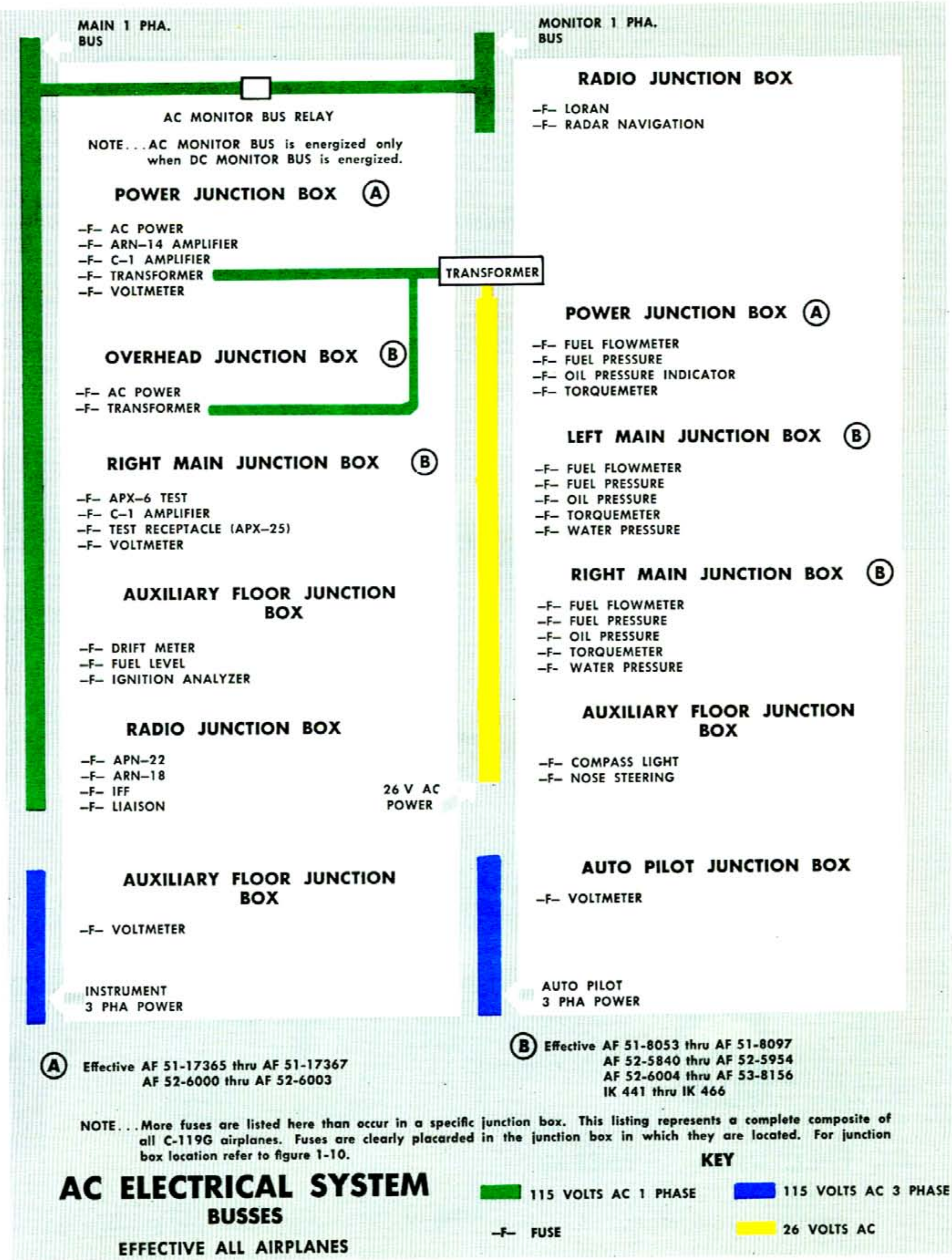

MAIN 1 PHA. BUS
MONITOR 1 PHA. BUS
AC MONITOR BUS RELAY
NOTE...AC MONITOR BUS is energized only when DC MONITOR BUS is energized.
POWER JUNCTION BOX (A)
–F– AC POWER
–F– ARN–14 AMPLIFIER
–F– C–1 AMPLIFIER
–F– TRANSFORMER
–F– VOLTMETER
TRANSFORMER
OVERHEAD JUNCTION BOX (B)
–F– AC POWER
–F– TRANSFORMER
RIGHT MAIN JUNCTION BOX (B)
–F– APX–6 TEST
–F– C–1 AMPLIFIER
–F– TEST RECEPTACLE (APX–25)
–F– VOLTMETER
AUXILIARY FLOOR JUNCTION BOX
–F– DRIFT METER
–F– FUEL LEVEL
–F– IGNITION ANALYZER
RADIO JUNCTION BOX
–F– APN–22
–F– ARN–18
–F– IFF
–F– LIAISON
26 V AC POWER
AUXILIARY FLOOR JUNCTION BOX
–F– VOLTMETER
INSTRUMENT 3 PHA POWER
RADIO JUNCTION BOX
–F– LORAN
–F– RADAR NAVIGATION
POWER JUNCTION BOX (A)
–F– FUEL FLOWMETER
–F– FUEL PRESSURE
–F– OIL PRESSURE INDICATOR
–F– TORQUEMETER
LEFT MAIN JUNCTION BOX (B)
–F– FUEL FLOWMETER
–F– FUEL PRESSURE
–F– OIL PRESSURE
–F– TORQUEMETER
–F– WATER PRESSURE
RIGHT MAIN JUNCTION BOX (B)
–F– FUEL FLOWMETER
–F– FUEL PRESSURE
–F– OIL PRESSURE
–F– TORQUEMETER
–F– WATER PRESSURE
AUXILIARY FLOOR JUNCTION BOX
–F– COMPASS LIGHT
–F– NOSE STEERING
AUTO PILOT JUNCTION BOX
–F– VOLTMETER
AUTO PILOT 3 PHA POWER
(A) Effective AF 51-17365 thru AF 51-17367
AF 52-6000 thru AF 52-6003
(B) Effective AF 51-8053 thru AF 51-8097
AF 52-5840 thru AF 52-5954
AF 52-6004 thru AF 53-8156
IK 441 thru IK 466
NOTE...More fuses are listed here than occur in a specific junction box. This listing represents a complete composite of all C-119G airplanes. Fuses are clearly placarded in the junction box in which they are located. For junction box location refer to figure 1-10.
AC ELECTRICAL SYSTEM
BUSSES
EFFECTIVE ALL AIRPLANES
KEY
115 VOLTS AC 1 PHASE
115 VOLTS AC 3 PHASE
–F– FUSE
26 VOLTS AC

Figure 1–13B

verter is used independently to supply the power requirements of that equipment. However, should failure of the main single-phase inverter occur, the output of the spare single-phase inverter is automatically switched to assume the airplane's ac power load and the AN/APS-42 equipment is disconnected.

SINGLE PHASE TRANSFORMER.

A transformer located in the power junction box on some airplanes and in the overhead junction box on other airplanes reduces part of the single-phase inverter output to 26 volts for operation of the torque pressure, fuel, water and oil pressure indication systems.

THREE-PHASE INSTRUMENT INVERTERS.

Power to operate the pilot's attitude and directional gyros is provided by two 100 volt-ampere, three-phase, 115-volt, 400 cycle inverters located on the auxiliary floor. The inverters are designated main and spare. In normal operation the main instrument inverter is used; should it become inoperative, an automatic changeover unit switches the spare instrument inverter into the system. When a complete generator power failure occurs, a switch is provided to permit energization of the spare instrument inverter by battery. This assures that the pilot's attitude and directional gyros remain operative. On airplanes equipped with a B-6 driftmeter, the spare instrument inverter is used independently to supply the power requirements of that equipment. However, should a failure of the main instrument inverter occur, the output of the spare instrument inverter is automatically switched from the driftmeter and used to energize the pilot's attitude and directional gyros.

AUTOMATIC PILOT INVERTER.

A 500 volt-ampere, three-phase, 115-volt, 400 cycle inverter, located on a shelf in the top rear of the cargo compartment supplies power to energize the automatic pilot system.

AC MONITOR BUS.

An ac monitor bus arrangement is incorporated into the single-phase ac power system. This ac monitor bus is energized only when its counterpart, the dc monitor bus, is energized. Should the output of an engine-driven generator be excluded from the bus for any reason, the dc monitor bus, and correspondingly, the ac monitor bus are deenergized. Power to both monitor busses may be restored by use of the monitor bus switch. With no voltage on the ac monitor bus, the APN-12 and APN-70 equipment is inoperative.

INVERTER SWITCH AND WARNING LIGHTS.

A three-position switch (17, figure 1–6) on the overhead panel controls the selection of the main or spare single phase inverter. In the center OFF position of the switch, neither inverter is energized. The MAIN position permits the main inverter to supply the power for the ac system. Should the main inverter fail while the switch is in the MAIN position, an inverter failure warning light (25, figure 1–22) on the instrument panel labelled "MAIN" will come on. At the same time an automatic changeover relay will function to permit the spare inverter to assume the responsibility of supplying the ac system.

Note

> When the inverter selector switch is in MAIN and the inverter fails, a second inverter failure warning light on the instrument panel labelled "BOTH" (25, figure 1–22) will momentarily light until the automatic changeover is accomplished.

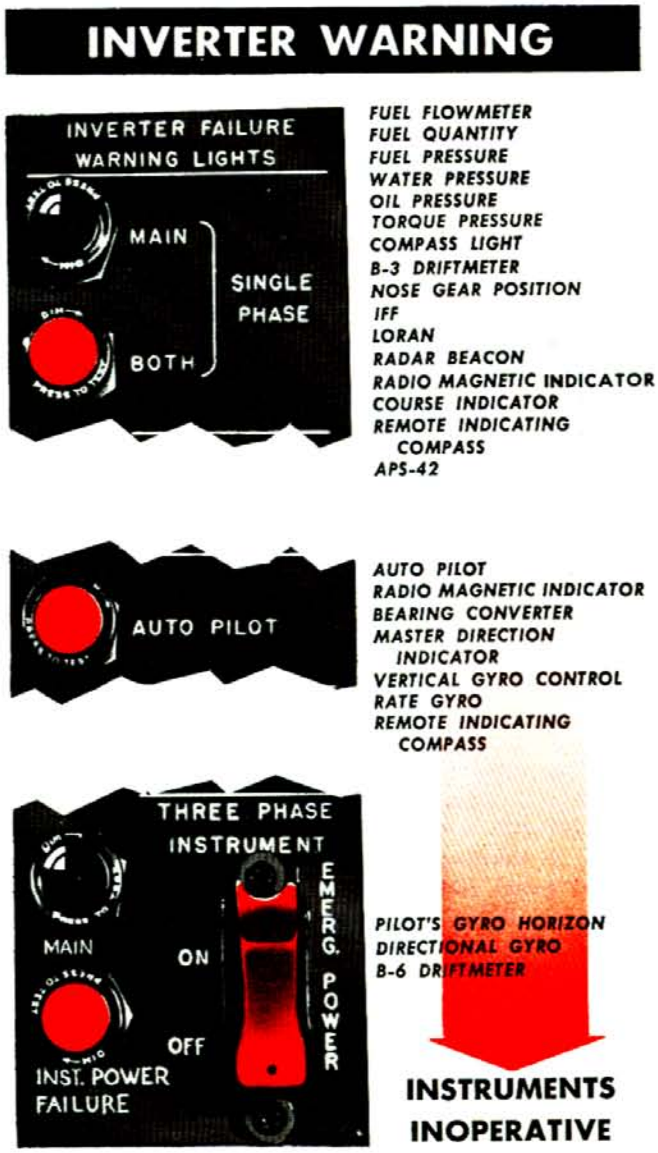


Figure 1–14

The SPARE position of the inverter control switch is used to manually connect the spare inverter to the system should the automattic changeover relay fail to operate. If the switch is in the SPARE position and the main inverter is not operating, the main inverter warning light will glow. Should the spare inverter fail while the selector switch is turned to SPARE, the automatic changeover relay will not allow an automatic return to the main inverter even though the main inverter is operative and the BOTH light will glow. Manual operation of the selector switch is required to place the main inverter in the ac system as a source of power; the MAIN and BOTH inverter failure lights will then go out. If both inverters fail, this condition is indicated by the glowing of the BOTH inverter failure warning light on the instrument panel. With the loss of power in the ac system certain equipment and instruments become inoperative as shown in figure 1–14.

Note

The fuel quantity indicators are so constructed that they will continue to indicate the conditions prevailing in their respective systems at the time of electrical failure.

AN/APS-42 INVERTER SWITCH.

On airplanes equipped with AN/APS-42 equipment, an OFF-ON switch is provided on the navigator's instrument panel to energize the spare single-phase inverter, and to supply ac power to the AN/APS-42 equipment. Should the main inverter fail, the automatic changeover relay, regardless of the setting of the spare inverter switch, will cut off the ac power to the AN/APS-42 equipment and direct the output of the spare inverter to the ac circuits of the airplane normally supplied by the main inverter. AN/APS-42 power is also automatically cut off when an engine generator failure occurs.

AUTOMATIC PILOT INVERTER SWITCH AND WARNING LIGHT.

A two-position automatic pilot inverter OFF-ON switch (18, figure 1–6) on the overhead panel controls the inverter which energizes the automatic pilot system. A warning light (26, figure 1–22) on the instrument panel indicates failure of the automatic pilot inverter. When the automatic pilot inverter is inoperative, the instruments as shown in figure 1–4 are also inoperative.

Note

The pilot's set of the above instruments is energized by the instrument inverters and, being separate from the automatic pilot system, is not affected by an inoperative condition of the automatic pilot inverter.

INSTRUMENT INVERTER SWITCH AND WARNING LIGHTS.

A three-position switch (19, figure 1-6) on the overhead panel controls the selection of two instrument inverters located on the auxiliary floor. In the center OFF position of the switch, neither instrument inverter is energized. These inverters are utilized to supply three phase ac power to the attitude and direction gyro indicators. One inverter is utilized as the main; the other, as a spare. The MAIN position of the switch is used during normal operation and permits the main inverter to supply the power for the instruments. Should the main inverter fail while the switch is in the MAIN position, an instrument inverter failure warning light (28, figure 1-22) on the instrument panel labelled MAIN will come on. At the same time an automatic changeover relay will function to permit the spare inverter to assume the responsibility of supplying power to the gyro instruments.

Note

When the instrument inverter switch is in MAIN and the inverter fails, a second instrument inverter failure warning light (28, figure 1-22) on the instrument panel labelled INSTRUMENT POWER FAILURE will momentarily light until the automatic changeover is accomplished.

The SPARE position of the instrument inverter switch is used to manually connect the spare inverter to the system should the automatic changeover relay fail to operate. If the switch is in the SPARE position and the main inverter is inoperative, the MAIN instrument inverter warning light will glow. Should the spare inverter fail while the selector switch is turned to SPARE, the automatic changeover relay will not allow an automatic return to the main inverter even though the main inverter is operative, and the INSTRUMENT POWER FAILURE light will glow. Manual operation of the switch to MAIN is required to place the main inverter in the system as a source of power; the MAIN and INSTRUMENT POWER FAILURE lights will then go out. If both inverters fail, this condition is indicated by the glowing of the INSTRUMENT POWER FAILURE warning light on the instrument panel and the instruments shown on figure 1-14 become inoperative. On later airplanes, operation of the spare instrument inverter may also be controlled by the navigator's driftmeter switch. When the main instrument inverter is operating and the driftmeter switch is turned to ON, the spare instrument inverter is also energized and provides additional ac current for operation of the driftmeter. Should the main instrument inverter fail while the driftmeter is in operation, the spare instrument inverter will continue to operate. However, a relay will function to automatically shut off current to the driftmeter, thus assuring sufficient output from the spare inverter for the pilot's directional gyro and attitude indicators.

Note

On airplanes IK-441 and 442 the warning light indicating failure of both instrument inverters is labelled BOTH. Illumination of the light indicates a loss of three-phase power for operation of the pilot's attitude and directional gyros.

AC VOLTMETERS AND DC VOLTAGE SELECTOR SWITCH.

An ac voltmeter and voltage selector switch (29, figure 1-22) are installed on the instrument panel to provide inverter voltage readings. The five positions and functions of the selector switch are as follows:

OFF—Voltmeter inoperative.

SINGLE PHASE—The voltmeter will indicate the voltage output of whichever single phase inverter is operating.

THREE PHASE—The voltmeter will indicate the ac voltage output of whichever three phase instrument inverter is operating.

SPARE INSTRUMENT INVERTER—The voltmeter will indicate the ac voltage output of the spare instrument inverter.

AUTO PILOT INVERTER—The voltmeter will indicate the ac voltage of the auto pilot inverter.

HYDRAULIC POWER SUPPLY SYSTEM.

Hydraulic pressure is used as the medium through which the following components are operated: landing gear, wing flaps, elevator lock, nose wheel steering, and main wheel brake systems. The complete hydraulic system on this airplane can be visualized as having a power supply system and individual systems that are actuated hydraulically (the hydraulic function of the individual systems will be covered under each system). The hydraulic power system can be thought of as three separate systems although they are actually interconnected. The three systems are the supply system, the normal power system, and the emergency power system.

SUPPLY SYSTEM.

The supply system consists of a main reservoir, having a capacity of 9.3 gallons and which is located on the floor of the auxiliary equipment compartment, and an emergency reservoir, having a capacity of 2.3 gallons, which is located in the cargo compartment ceiling. The emergency reservoir is filled from the main reservoir; however, being an independent reservoir, it will remain filled, even though the main reservoir is depleted, until such time as it is depleted by operation of the emergency system. The main reservoir is provided with a filler point and oil level is determined through the use of a sight gage on the tank. Both reservoirs are vented to the atmosphere. The total capacity of the hydraulic system is approximately 16.5 gallons. Refer to the Servicing Diagram (Figure 1-25) for hydraulic oil specification.

NORMAL POWER SYSTEM.

The normal power system consists of an engine-driven pump on each engine, which draws oil from the main reservoir and supplies it under pressure through a filter to the pressure regulator. The pressure regulator automatically maintains an operating pressure for the operation of the landing gear and the wing flaps which are high pressure systems. A pressure reducing valve tapped off the high pressure line reduces the pressure to the low pressure system for operation of the elevator lock, nose wheel steering, and main wheel brakes. A high pressure accumulator is provided to supply the initial charge when any of the hydraulic components are operated. Relief valves are utilized throughout the system to guard against any abnormal pressures in the hydraulic lines. In addition, a manually operated pressure release valve permits depressurization of the system in an emergency. High and low pressure gages are located on the instrument panel and on the hydraulic emergency control panel on the auxiliary floor.

FUEL, OIL AND HYDRAULIC SHUT-OFF SWITCHES.

Hydraulic system shut-off valves located in the supply line at each engine firewall are electrically actuated by the fuel, oil and hydraulic shut-off switches (22, 31, figure 1-6). When either of the switches is placed in the NORM position, with the switch down and switch guard closed, the corresponding 28-volt dc valve opens and will remain open until the switch is moved

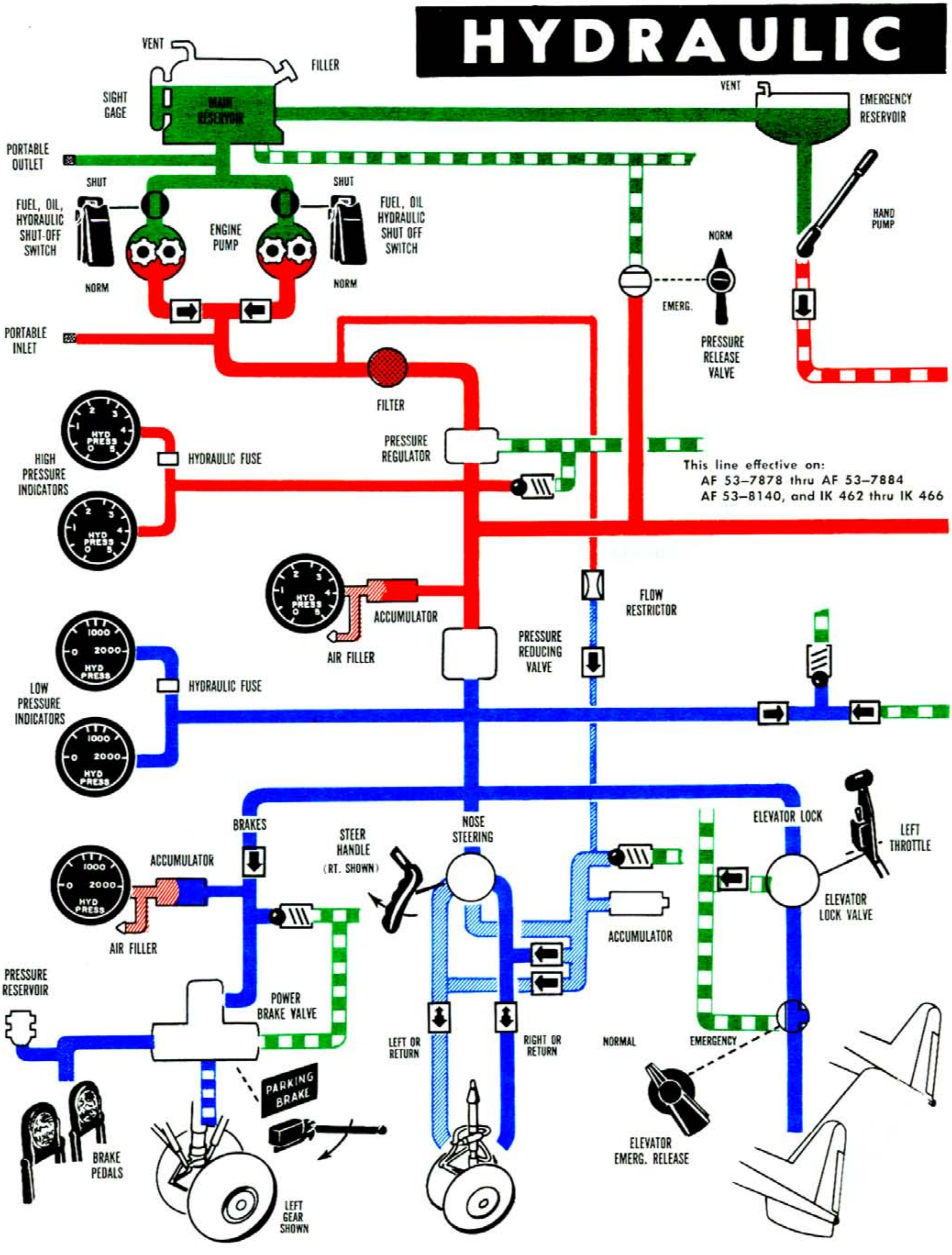


Figure 1-15 (Sheet 1 of 2 sheets) Hydraulic System Schematic

SYSTEM

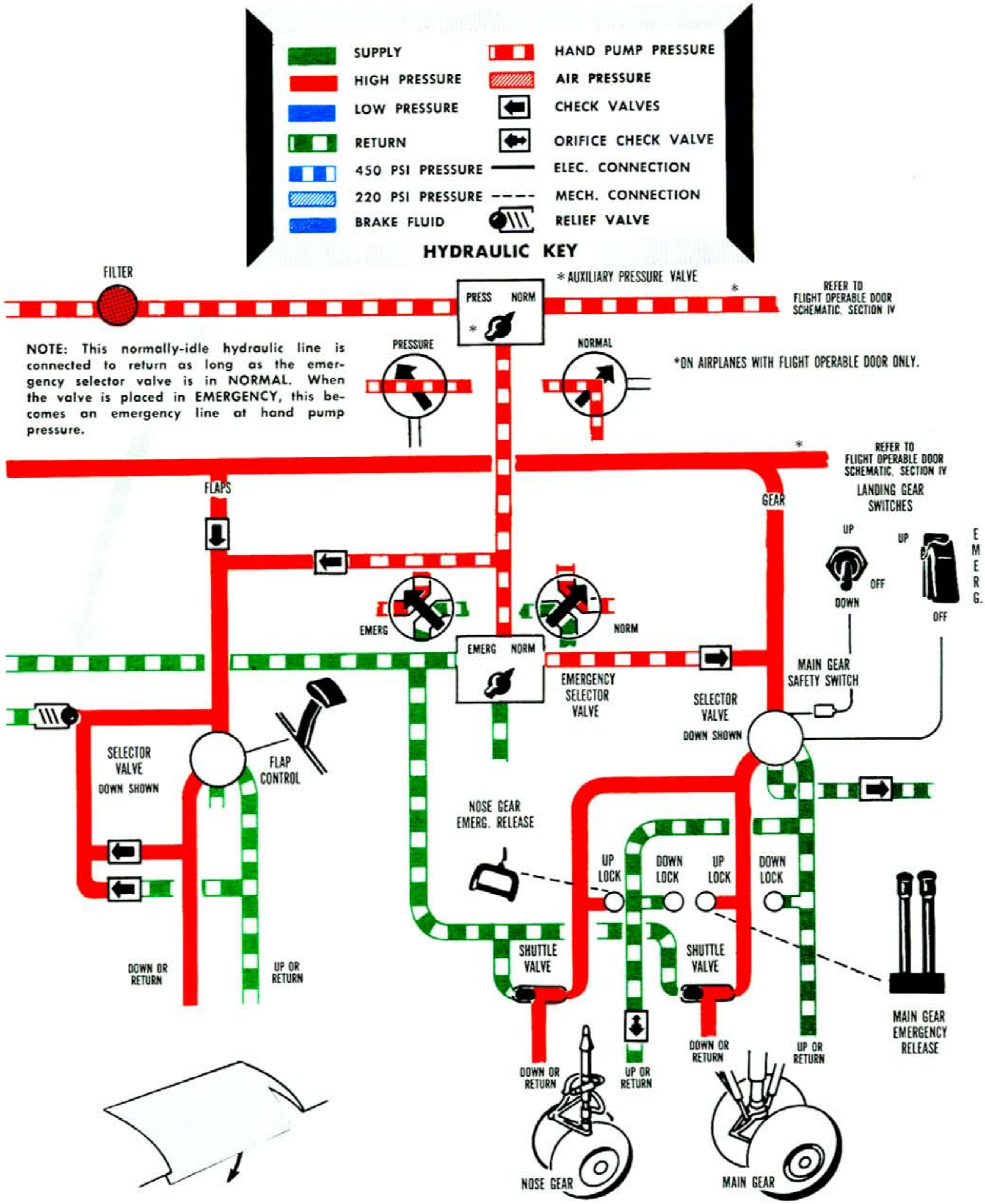


Figure 1–15. (Sheet 2 of 2 sheets) Hydraulic System Schematic

to the SHUT position with the switch up and switch guard open.

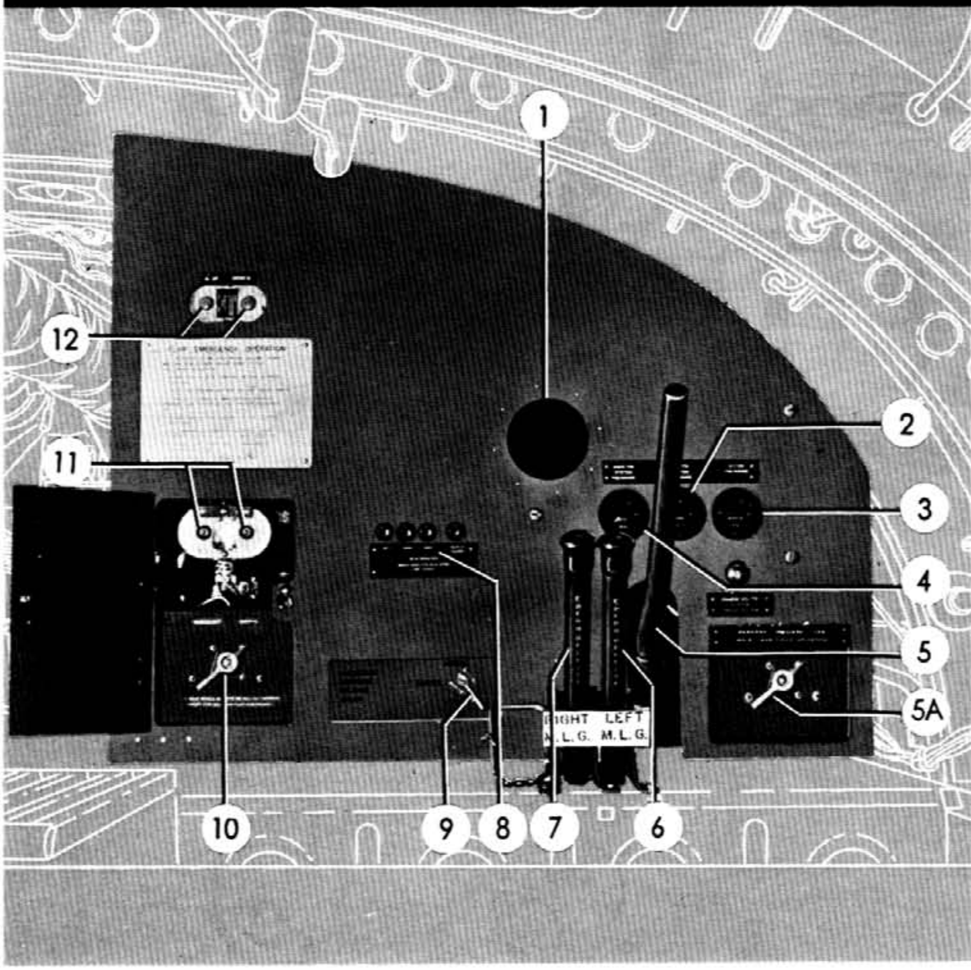
EMERGENCY PRESSURE RELEASE VALVE (SOME AIRPLANES).

A manually-operated valve (9, figure 1-16), installed on the hydraulic emergency control panel, provides a means of depressurizing the system in the event excessive hydraulic pressures or undesirable cycling of the system should occur. When the valve is turned to the EMERGENCY position, engine-pump pressure is diverted directly to the main reservoir. However, accumulator pressure is maintained in the brake system regardless of the position of the pressure release valve. Returning the valve to the NORMAL position will restore pump pressure to the system and close the line to the reservoir.

HYDRAULIC SYSTEM PORTABLE PUMP.

To eliminate the necessity of engine run-up during ground maintenance, a portable hydraulic pump may be connected to the hydraulic external power lines which connect to the supply and pressure lines of the main system. The external power lines are located under a cover plate on the left side of the fuselage. Refer to Servicing Diagram (Figure 1-25).

HYDRAULIC EMERGENCY CONTROL PANEL



1. HYDRAULIC OIL LEVEL GAGE ACCESS
2. HYDRAULIC LOW PRESSURE GAGE
3. HYDRAULIC ACCUMULATOR AIR PRESSURE GAGE
4. HYDRAULIC HIGH PRESSURE GAGE
5. HYDRAULIC EMERGENCY HAND PUMP

5 A. FLIGHT OPERABLE DOOR AUXILIARY PRESSURE VALVE

6. LEFT MAIN LANDING GEAR EMERGENCY UP-LOCK RELEASE HANDLE
7. RIGHT MAIN LANDING GEAR EMERGENCY UP-LOCK RELEASE HANDLE
8. LANDING GEAR INDICATOR LIGHTS
9. EMERGENCY PRESSURE RELEASE VALVE
10. HYDRAULIC SYSTEM EMERGENCY SELECTOR VALVE
11. LANDING GEAR SELECTOR VALVE MANUAL CONTROL BUTTONS
12. WING FLAP SELECTOR VALVE MANUAL CONTROL BUTTONS

Figure 1-16

EMERGENCY POWER SYSTEM.

In the event of engine-driven pump failure or hydraulic line leakage an emergency system, with separate hydraulic lines, is provided for operation of the landing gear and wing flaps. This system consists of an emergency hand pump which draws its supply from the emergency reservoir, and an emergency selector valve which governs the direction of the emergency oil flow.

EMERGENCY HAND PUMP.

The emergency hand pump (5, figure 1-16) is located on the hydraulic emergency control panel on the auxiliary floor. Pressure resulting from its use is directed according to the positioning of the emergency selector valve.

EMERGENCY SELECTOR VALVE.

A manually-operated emergency selector valve (10, figure 1-16) is located behind the cover plate on the hydraulic emergency control panel on the auxiliary floor. This valve is provided with EMERGENCY and NORMAL positions. Upon operation of the hand pump with the emergency selector valve in the EMERGENCY position, pressure is directed through the emergency down lines to the landing gear actuators and the selected part of the wing flaps selector valve. The wing flaps selector valve may be operated either electrically or manually.

Note

Although the emergency selector valve does not control the hand pump pressure available at the wing flaps selector valve, the emergency selector valve must be placed in the EMERGENCY position during the emergency operation of the wing flaps. This action is necessary to retain hand pump pressure wholly within the emergency lines and prevent its dissipation throughout the hydraulic system.

The emergency selector valve is retained in the NORMAL position when the hydraulic power system is functioning normally. Operation of the hand pump with the emergency selector valve in NORMAL will build up pressure in the complete hydraulic system.

FLIGHT CONTROLS.

AILERON CONTROLS.

Control wheels for pilot and copilot and associated cable assembly are used to effect mechanical aileron control. Full deflection of the ailerons is obtained by 150° rotation of the control wheels each side of the neutral position. Positive stops are provided so that travel beyond prescribed limits is impossible. An automatic pilot clutch disengage button and a microphone control button are installed on the outboard side of each control wheel. Flettner tabs are provided on each inboard aileron to assist movement of the aileron surfaces, thereby reducing control wheel loads during flight. The aileron trim tab, designed to provide aileron adjustment affecting lateral balance of the airplane, is located on the right inboard aileron.

AILERON TRIM TAB CONTROL SWITCH AND INDICATOR.

A three-position aileron trim tab switch (9, figure 1-5) is provided on the pilot's switch panel. The LEFT WING DOWN and RIGHT WING DOWN positions are momentary and result in airplane attitudes as the names imply. The center OFF position of the switch will hold the trim tab position last selected. Movement of the trim tab switch electrically operates an actuator located in the leading edge of the right inboard aileron. The actuator, through mechanical linkage, positions the trim tab. Also incorporated in the actuator is a trim tab position transmitter which generates the signal reflected on the trim tab indicator (10, figure 1-5) in degrees of deflection.

ELEVATOR CONTROLS.

A single elevator surface located at the trailing edge of the stabilizer and extending the entire span between the twin booms, is the means by which the airplane is controlled about its lateral axis. Fore and aft movement of the control columns controls the positioning of the elevator. Full-up elevator is obtained by approximately 10 inches of aft control column movement, and full-down elevator by approximately 7 inches of forward control column movement. Positive stops are provided so that excessive travel of the elevator control system beyond prescribed limits is impossible. An elevator spring tab, operating automatically with control column movement, assists elevator movement during flight and lessens pilot control effort necessary for maneuvering.

ELEVATOR TRIM TAB CONTROL AND INDICATOR.

The elevator trim tab is installed in the left side of the elevator trailing edge and is mechanically actuated by two interconnected control wheels (5, figure 1-4) mounted on opposite sides of the control quadrant. Mechanically geared to the trim tab control wheels, the elevator trim tab indicators (5, figure 1-4) are an integral part of the control wheels and indicate position of the tab in degrees of deflection.

ELEVATOR GUST LOCK.

The hydraulic elevator gust lock is provided so that the elevator is locked during reverse pitch operation or may be engaged during taxiing operations under gusty conditions. The elevator lock is a hydraulic operated lock which is controlled by movement of the left throttle. Whenever the left engine throttle is moved into the reverse range, a hydraulic selector valve is electrically positioned so that hydraulic pressure may accomplish the locking action. The elevator will remain locked until the throttle is moved forward to a position at which the engines develop a manifold pressure of approximately 38 in. Hg. This delayed action on the unlocking cycle is provided so that the lock may be applied and used during taxi operations. To accomplish this locking operation for taxiing, the left throttle should be moved into the

PILOT'S STATION

1. HAND MICROPHONE
2. NOSE WHEEL STEERING HANDLE
3. JUMP SIGNAL CONTROL PANEL
4. NOSE WHEEL POSITION INDICATOR
5. FLIGHT CONTROL LOCK CABLE AND HOOK
6. MICROPHONE CONTROL BUTTON
7. AUTOMATIC PILOT CLUTCH DISENGAGE BUTTON
8. AUTOMATIC PILOT EMERGENCY DISCONNECT HANDLE AND ELEVATOR LOCK RELEASE
9. FLIGHT CONTROLS LOCK PLUNGER
10. FLIGHT CONTROLS LOCK ACCESS DOOR
11. OXYGEN FLOW INDICATOR
12. OXYGEN PRESSURE GAGE
13. SEAT ADJUSTMENT LEVER (VERTICAL)
14. SEAT ADJUSTMENT LEVER (RECLINING)
15. STANDBY LOUDSPEAKER
16. EQUIPMENT DROP RELEASE HANDLE
17. PARKING BRAKE
18. INTERPHONE AND FILTER CONTROL PANELS
19. SIDE WINDOW LOCK

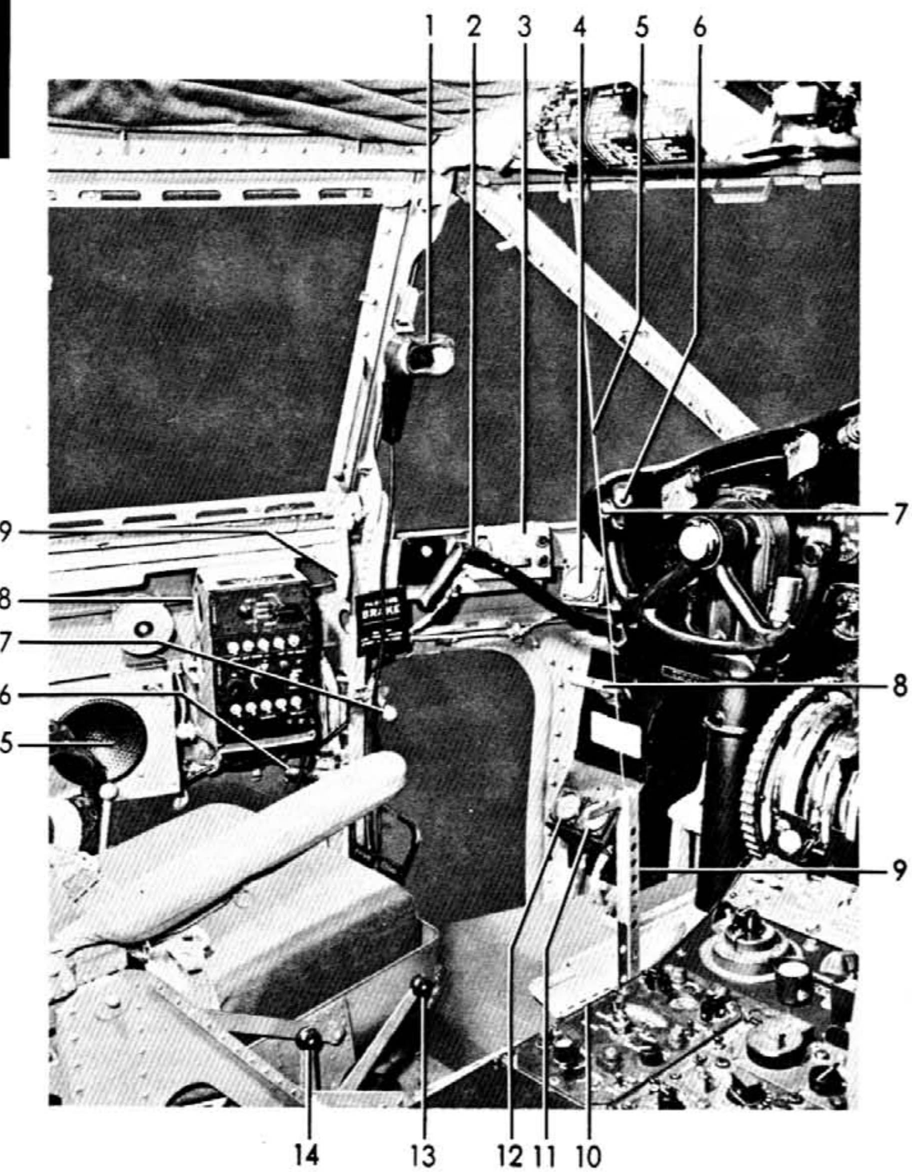


Figure 1-17

reverse pitch range momentarily and then advanced to forward thrust.

Note

A safety switch on the nose landing gear permits the elevator to be locked only when the gear is extended and its shock strut compressed.

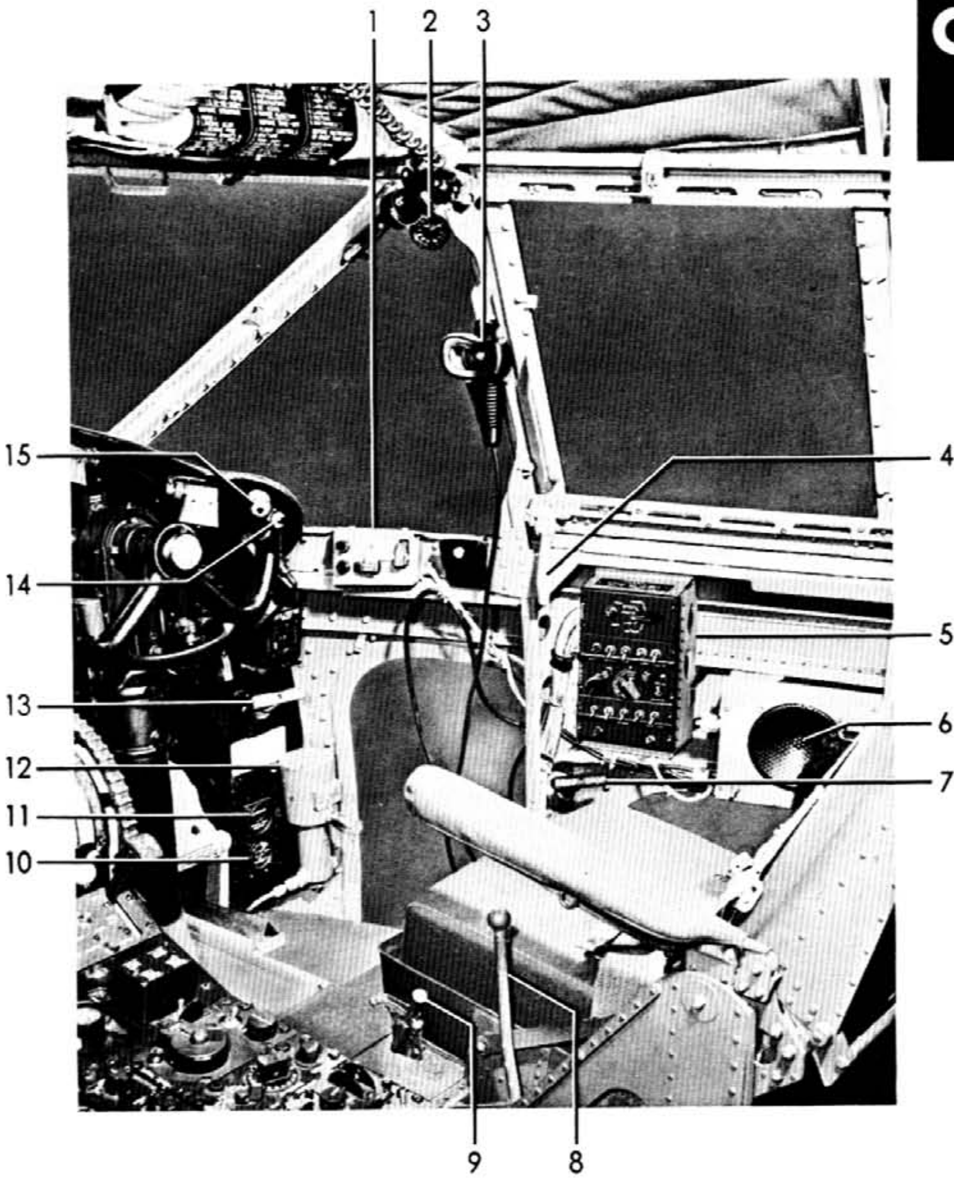
ELEVATOR GUST LOCK EMERGENCY RELEASE VALVE.

An elevator gust lock emergency release valve, with NORMAL and EMERGENCY positions, is installed overhead just to the right of the monorail in the approximate center of the cargo compartment. When turned to EMERGENCY position, the hydraulic valve is mechanically positioned to assure complete release of all hydraulic gust lock pressure. The automatic pilot disconnect handles at the pilot's and copilot's stations are cable connected to the elevator lock release valve and when pulled, automatically position the valve to EMERGENCY. However, manual movement of the valve does not disconnect the automatic pilot.

CAUTION

When the elevator gust lock release valve is turned to EMERGENCY, the elevator gust lock is inoperative until the valve is repositioned to NORMAL.

COPILOT'S STATION



1. JUMP SIGNAL CONTROL PANEL
2. FREE AIR TEMPERATURE GAGE
3. HAND MICROPHONE
4. SIDE WINDOW LOCK
5. INTERPHONE AND FILTER CONTROL PANELS
6. STANDBY LOUDSPEAKER
7. EQUIPMENT DROP RELEASE HANDLE
8. SEAT ADJUSTMENT LEVER (FORWARD, AFT AND LATERAL)
9. SHOULDER HARNESS INERTIA REEL LOCK CONTROL
10. OXYGEN PRESSURE GAGE
11. OXYGEN FLOW INDICATOR
12. HYDRAULIC BRAKE PEDAL PRESSURE RESERVOIR
13. AUTOMATIC PILOT EMERGENCY DISCONNECT HANDLE AND ELEVATOR LOCK RELEASE
14. AUTOMATIC PILOT CLUTCH DISENGAGE BUTTON
15. INTERPHONE MICROPHONE BUTTON

Figure 1-18

RUDDERS.

Dual rudders hinged to the trailing edge of each fin are the means by which the airplane is controlled about its vertical axis. Each set of rudder pedals can be adjusted to accommodate pilots of different stature by use of the adjustment wheels located below the instrument panel. Positive stops in the system prevent excessive rudder travel beyond prescribed limits. Rudder spring tabs, provided on the bottom trailing edge of each rudder and interconnected with rudder pedals, automatically serve to lessen the effort necessary to move the rudder surface during flight.

RUDDER TRIM TAB CONTROL AND INDICATOR.

Trim tabs installed on the trailing edge of each rudder surface immediately above the spring tabs provide a means of trimming the airplane directionally. Control of the rudder trim tabs is electrically maintained by a three-position switch (7, figure 1-5) on the pilot's switch panel. The switch positions, NOSE LEFT and NOSE RIGHT are spring-loaded and provide directional trim of the airplane as their names imply. The center OFF position will hold the last selected position on the rudder trim tabs. An electrical indicator (8, figure 1-5), mounted on the pedestal adjacent to the rudder trim tab switch, gives a visual indication of the trim tab position in degrees of deflection.

FLIGHT CONTROLS LOCK.

A cable-operated flight controls lock (5, 9, figure 1-17) is provided to prevent movement of the control surfaces when the airplane is parked. It is a manually operated, cable controlled system actuated by a plunger handle which is recessed in the floor at the pilot's station. When this handle is extended to the locked position and the cable within it is attached to the hook at the top of the canopy, it presents a conspicuous arrangement that reduces the possibility of the pilot attempting flight operations without first unlocking the flight controls.

CAUTION

When unlocking the flight controls make certain that the plunger handle is completely recessed in the floor, thus assuring that the flight controls lock is disengaged.

The flight controls lock will only engage when the rudder and aileron controls are in neutral. The control column should be positioned forward. Locking may be completed by pulling up on the plunger handle while applying a slight back and forth movement to the rudder and aileron controls. The spring-loaded cable of the plunger handle is then attached to a receptacle provided at the top of the pilot's windshield. Positive lock of the surfaces may be checked by applying pressure to the control wheel and rudder pedals. The elevator control will have a small degree of movement. The flight controls lock is disengaged by unfastening the cable and restraining the cable as it returns within the plunger. Do not allow the flight controls lock cable to snap back into plunger as the cable, under spring tension, may jump off the reel.

Note

Do not engage the flight controls lock when the hydraulic gust lock is engaged as damage to the cable control system of the elevator lock may result.

WING FLAPS.

Wing flaps, of the slotted type, are installed in the trailing edge of each wing between the inboard edge of the ailerons and the outboard side of the nacelle, and between the inboard side of the nacelle and the side of the fuselage. The wing flaps are hydraulically operated by the high pressure hydraulic system.

WING FLAP LEVER AND INDICATOR.

The wing flap lever (6, figure 1-5), with the positions UP, TAKE-OFF, and DOWN, is located on the pilots' switch panel. When placed in the UP or DOWN position, it electrically (28 volts dc) positions the wing flaps hydraulic selector valve and permits hydraulic pressure to accomplish the movement of the wing flap surfaces to either the up position or to 40° down position. When placed in TAKE-OFF position, an intermediate limit switch is energized which, when the flaps reach 14° down, serves to again position the hydraulic selector valve. The valve then retains pressure in both the up and down hydraulic lines, and holds the flaps in the 14° intermediate position. The position of the flaps is shown by an indicator (35, figure 1-22) on the instrument panel.

WING FLAP SELECTOR VALVE MANUAL CONTROL BUTTONS.

The wing flap selector valve which is controlled electrically by the flap control lever may also be controlled by the use of manual control buttons with UP and DOWN positions (12, figure 1-16). These manual control buttons are located on the hydraulic emergency control panel on the auxiliary floor. In the event of electrical system failure these buttons may be utilized to control the wing flap movement.

EMERGENCY SELECTOR VALVE.

The hydraulic emergency selector valve (10, figure 1-16) and the hand pump are utilized to effect operation of the wing flaps in the event the main hydraulic power system fails. This valve is provided with EMERGENCY and NORMAL positions. Upon operation of the hand pump with the selector valve in the EMERGENCY position, pressure is directed through the emergency lines to the wing flaps selector valve; in conjunction with this operation, the wing flap selector valve may be operated either electrically or manually.

Note

Although the emergency selector valve does not control the hand pump pressure available to the wing flaps selector valve, it must be placed in the EMERGENCY position during the emergency operation of the wing flaps. This action is necessary to retain hand pump pressure wholly within the emergency lines and prevent its dissipation throughout the hydraulic system.

LANDING GEAR SYSTEM.

The landing gear on this airplane is of the tricycle-type with a nose gear which is steerable 60° in either direction and with dual wheels on each of the main gears. Retraction and extension of the landing gear is accomplished through hydraulically operated actuators. All three gears are operated simultaneously with both the retraction and extension cycle requiring approximately 11 seconds. The landing gear is normally operated through the high pressure hydraulic system, high pressure from the engine-driven pumps being directed through the landing gear selector valve to the proper side of the landing gear actuators. When a malfunction makes the normal hydraulic power system inoperative, the hydraulic emergency power system (hand pump) may be utilized to extend the landing gear.

LANDING GEAR GROUND LOCK.

Ground locking pins for main and nose landing gear, manually installed and removed, are a safety device to prevent inadvertent retraction during maintenance and

GREEN OFF, RED OFF

Landing gear up and locked, throttle forward beyond minimum cruising power.

GREEN OFF, RED ON

Landing gear in some intermediate position between up and locked and down and locked. Throttles forward beyond minimum cruising.

Landing Gear INDICATION

GREEN OFF, HORN STOPPED, RED ON

Landing gear up and locked or in some intermediate position, throttles below minimum cruising, warning horn disconnected.

GREEN OFF, HORN ON, RED ON

Landing gear up and locked or in some intermediate position, and throttles below minimum cruising.

GREEN ON, RED OFF

Landing gear down and locked regardless of throttle position.

Figure 1-19

ground handling. When not in use, the pins are carried in the left side of the forward cargo compartment just aft of the front entrance door.

LANDING GEAR SWITCH.

The landing gear switch (38, figure 1–22) is located on the instrument panel. This switch has UP, DOWN, and OFF positions and electrically positions the 28-volt dc hydraulic selector valve which allows the hydraulic pressure to actuate the up or down locks and extend or retract the gear for all normal operations. The center OFF position shuts off the pressure at the selector valve when the gear is up and locked. Safety switches are incorporated on the right main landing gear which render the UP position of the selector switch inoperative whenever the airplane is on the ground and the shock struts are compressed.

LANDING GEAR INDICATING SYSTEM.

Three green indicator lights and one red indicator light (37, figure 1–22), located on the instrument panel and on the hydraulic emergency control panel, comprise the landing gear position indicating system. A warning horn will blow at any time when the landing gear is other than down and locked and the throttles are below approximately 16-20 in. Hg.

A warning horn disconnect button (37, figure 1–22) is installed to permit this signal to be disconnected, but operation of the throttles above approximately 20-25 in. Hg. will automatically reset the signal circuit. Visual indication of landing gear position is provided by indicator lights as shown in figure 1–19.

LANDING GEAR EMERGENCY SWITCH.

A guarded landing gear emergency switch (36, figure 1–22) with UP and OFF positions is located on the instrument panel and marked "EMERGENCY." Since this emergency circuit bypasses the shock strut safety switches, it permits the pilot to retract the gear during the landing or take-off roll, should circumstances arise which demand this operation.

LANDING GEAR SELECTOR VALVE MANUAL CONTROL BUTTONS.

Two landing gear selector valve manual control buttons (11, figure 1–16) with UP and DOWN positions are located on the hydraulic emergency control panel on the auxiliary floor. These buttons, when depressed, provide manual positioning of the landing gear hydraulic selector valve, should electrical power failure render the normal landing gear switch inoperative.

LANDING GEAR EMERGENCY UP-LOCK RELEASE HANDLES.

The two main landing gear emergency up-lock release handles (6, 7, figure 1–16) are located just forward of the hydraulic emergency control panel on the auxiliary floor. If the normal hydraulic system should fail to extend the gear, these handles, when pushed release the lock holding the main landing gear in the up position, allowing it to fall free. The emergency up-lock release handle for the nose landing gear (11, figure 4–23) is located in the center of the cargo compartment forward bulkhead. Operate the hand pump to lock the gear in the down position.

EMERGENCY SELECTOR VALVE.

The hydraulic emergency selector valve (10, figure 1–16) and the emergency hand pump (5, figure 1–16) are utilized to effect emergency extension of the landing gear in the event the normal hydraulic power system fails. This valve is provided with EMERGENCY and NORMAL positions. Upon operation of the hand pump with the emergency valve in the EMERGENCY position, pressure is directed through the emergency lines to the landing gear shuttle valves.

STEERING SYSTEM.

The steering system is capable of turning the nose wheel 60° in either direction. With this amount of rotation, the airplane can be turned completely around within a 30 ft. radius. It is controlled by a steer handle located on the pilot's side of the crew compartment which actuates a switch and energizes a 28-volt dc steer valve. Steering is accomplished by use of hydraulic low pressure at the steer cylinder located on the nose gear shock strut. Installed in the return hydraulic lines are a relief valve and a steer system accumulator which maintain a 200 psi pressure on both sides of the steer cylinder for shimmy dampening. On airplanes AF 53-7878 through 53-7884, AF 53-8140, and IK-462 through 466, provisions are incorporated for restoring steering system pressure which may be dissipated through leaks by coupling the steering system to the main high pressure hydraulic system through a constantly bleeding restrictor. A safety switch on the nose gear breaks the steering control circuit when the airplane is airborne and the nose wheel reverts to the neutral position due to the centering action of the centering cam of the oleo strut. Should all hydraulic pressure in the steering system be lost, the nose wheel will trail in the centered position but all shimmy dampening will be lost.

CAUTION

Do not exceed limits given in TURNING LIMITATIONS, Section V.

NOSE GEAR STEERING

Note...

Refer to TURNING LIMITATIONS, Section V.

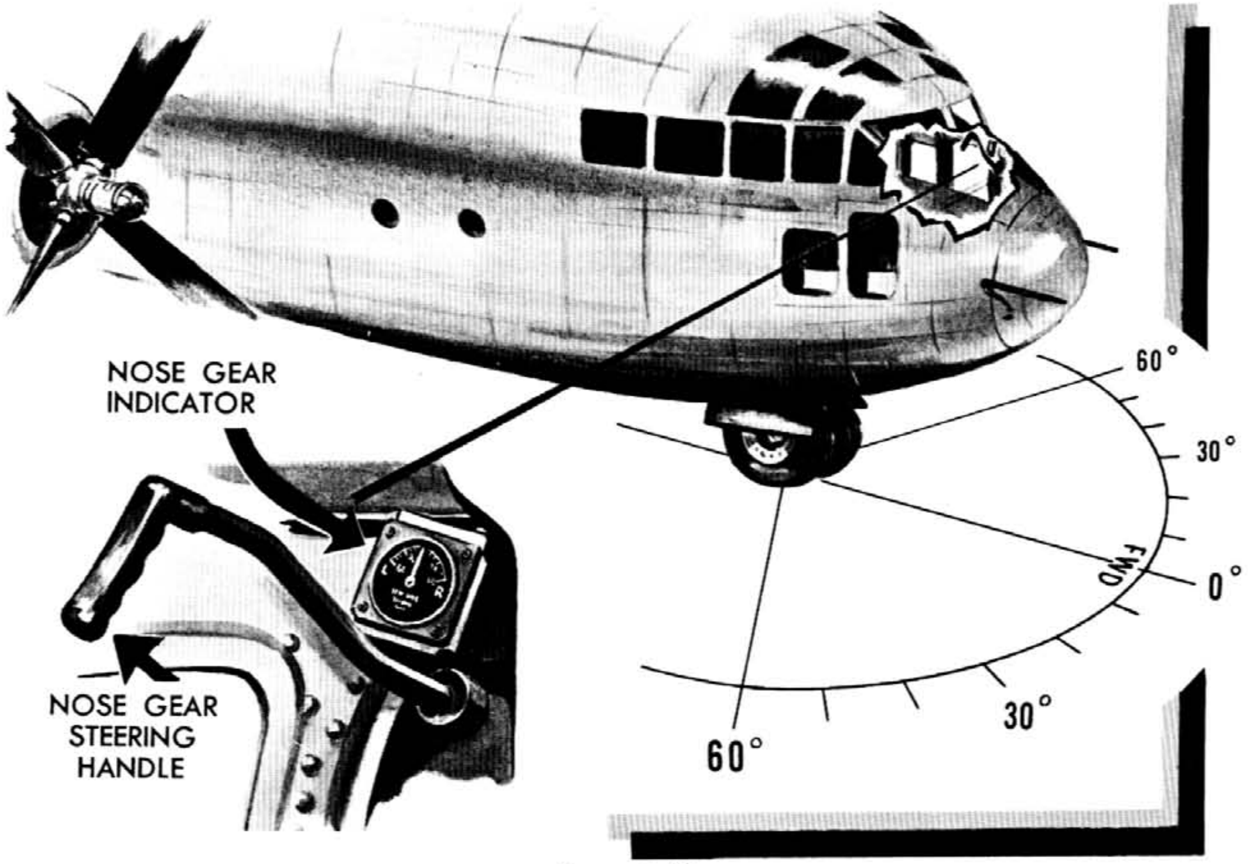


Figure 1-20

NOSE WHEEL STEERING HANDLE.

The nose wheel steering handle (2, figure 1-17) is located on the left side of the instrument panel in front of the pilot's seat. When turned, it electrically positions a 28-volt dc solenoid valve which allows hydraulic pressure to enter the proper side of the steer cylinder.

NOSE WHEEL POSITION INDICATOR.

A position indicator (4, figure 1-17) is located below the pilot's windshield side panel. The indicator shows the degree and direction of turn and functions in response to a signal received from a 26-volt ac transmitter mounted on the steer cylinder bracket. Movement of the nose gear strut by the hydraulic steer cylinder is transferred through mechanical linkage to the transmitter. The nose gear position indicator is inoperative if both single phase inverters or the 26-volt ac transformer malfunction.

BRAKE SYSTEM.

Wheel brakes provided on the main landing gear wheels are operated by the low pressure hydraulic system. The brakes are applied by depressing the toes of either the pilot's or copilot's pedals which control the operation of the power brake valve and ultimately determine the pressure available for braking at the rotor discs of the wheel brake assembly. The brake system is provided with a separate accumulator which, due to its pre-load air pressure of 450 psi, will maintain a brake system reserve pressure of 450 to 1150 psi in the event of a hydraulic power system failure. The air pressure gage for this accumulator is located in the lavatory compartment and indicates the pressure immediately available for the initial operation of the wheel brakes.

Note

A brake system accumulator air pressure gage reading of at least 600 psi will assure the pilot of braking pressure for at least one full application of the brakes.

AIR BRAKE Schematic

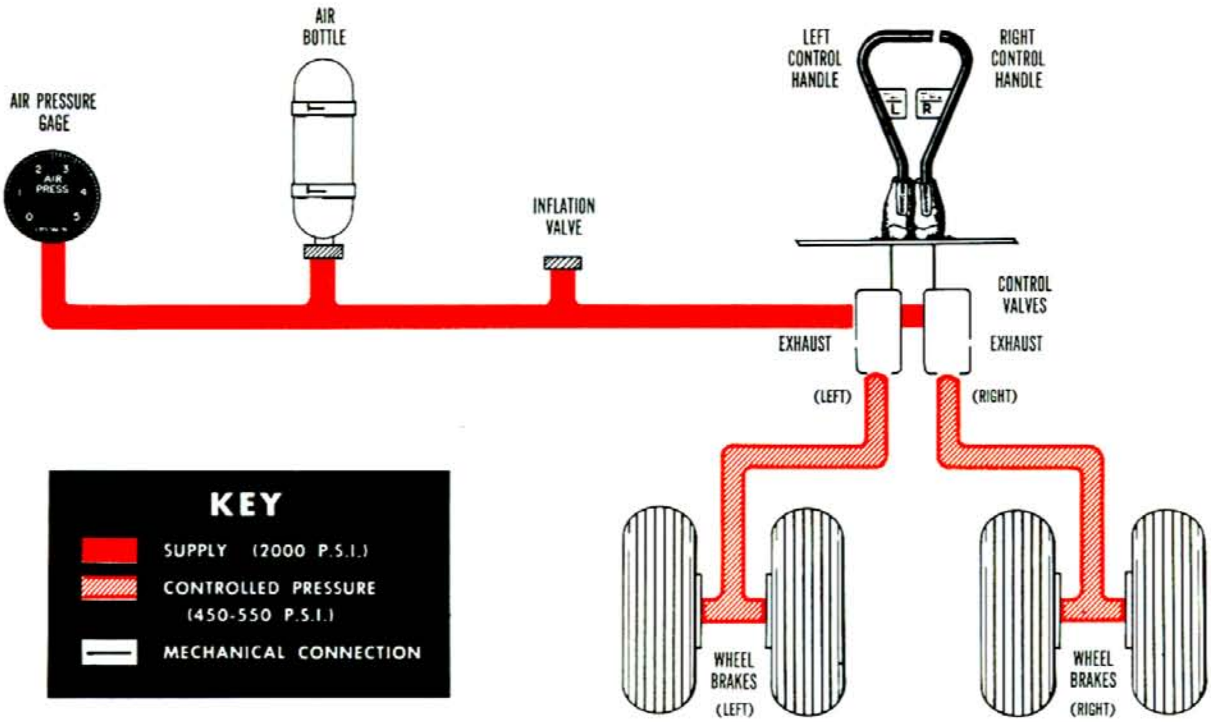


Figure 1-21

BRAKE PEDAL SYSTEM.

The brake pedal system is directly controlled by the brake pedals and is entirely independent of the main hydraulic system. The only function of the brake pedal system is to operate the power brake valve and thus determine the amount of pressure permitted to actuate the rotor discs of the wheel brakes. The brake pedal pressure reservoir (12, figure 1–18), located at the copilot's station, contains the fluid supply and maintains a constant pressure, regardless of altitude or atmospheric conditions, in the brake pedal system. When the toes of either the pilot's or copilot's pedals are depressed, the pressure applied by the hydraulic fluid is such that it opens the power brake valve, permitting hydraulic fluid to flow from the airplane's low pressure hydraulic system to the rotor discs of the wheel brakes. Variations in the pressure applied to the toe brakes control the positioning of the power brake valve and consequently, the hydraulic pressure being applied for actual braking action. The maximum pressure which the power brake valve will release for the braking action when the toe brakes are fully depressed, is 450 psi, regardless of the pressure prevailing in the hydraulic low pressure system.

Note

Correct brake pedal system operation is indicated when, upon depressing toe brakes, a momentary decrease in pressure is observed on the hydraulic low pressure gage. Brake pedal system fluid loss is indicated by the plunger on the pressure reservoir. If more then 3/4 inch of the rod remains exposed when the plunger is depressed, the reservoir should be replenished.

PARKING BRAKE.

The parking brake control handle (17, figure 1–17), located below the side window at the pilot's station, is provided with ON and OFF positions. The primary function of this handle is to apply brakes when the airplane is parked. However, the parking brake may also be used for emergency stops in the event malfunction of the brake pedal system makes the brake pedals inoperative. This is possible because a mechanical linkage, bypassing the independent brake pedal system, controls the flow of fluid from the main hydraulic system. When the control is in any intermediate position other than ON or OFF, the mechanical linkage positions the power brake valve, permitting a graduated portion of hydraulic pressure to be applied to the brakes.

CAUTION

When the control handle is in the ON position, brakes are full on. Do not use foot brakes when applying the parking brake, as excessive stress may be applied to the parking brake control system.

AIR BRAKE SYSTEM.

An air brake system (figure 1-21) independent of the brake pedal system and parking brake is provided on later airplanes. Two levers, installed on the floor just to the left of the pilot's station, control individual or simultaneous braking of the main wheels. Pressure to accomplish the braking action is released from an air bottle located just beneath the crew compartment floor and charged with 1600 to 2000 psi. The air gage is located to the left of the pilot inside the crew compartment and the filler point on the exterior of the airplane just below the pilot's lower side windows. When checking or recharging the system on these airplanes, the air gage is visible through the pilot's aft down vision window. On airplanes 53-7874 through 53-7884 and IK-458 through 466 a moisture drain valve is installed just below the air filler point. This drain should be operated in accordance with the placarded instructions each time the system is recharged in order to avoid the danger of brake malfunction due to moisture in the lines under low temperature conditions.

INSTRUMENTS.

ENGINE INSTRUMENTS.

The instrument panel is arranged with the engine instruments in the center visible to both the pilot and copilot. Refer to ENGINES, Section I, for a discussion of the engine instruments provided.

FLIGHT INSTRUMENTS.

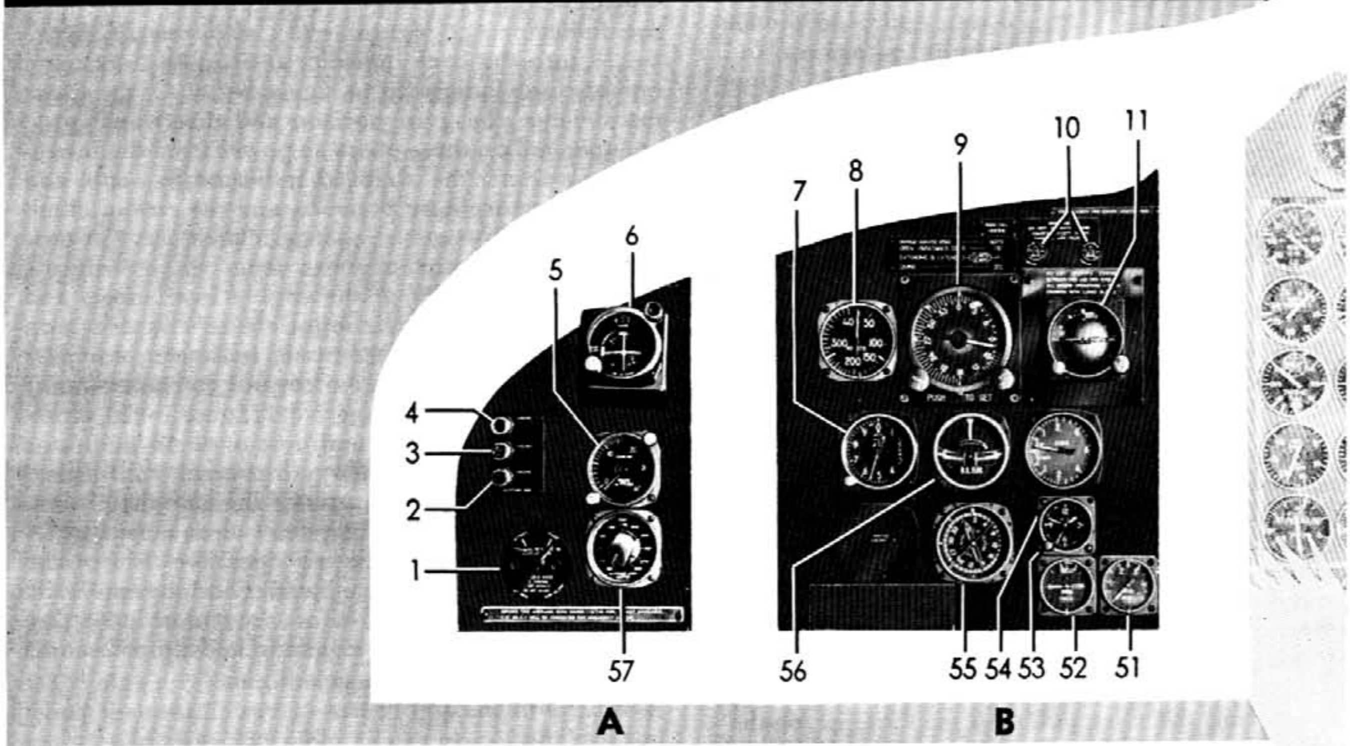
To the left of the engine instruments and directly in front of the pilot are the pilot's flight instruments which consist of an airspeed indicator (8, figure 1-22), directional gyro indicator (9, figure 1-22), gyro horizon indicator (11, figure 1-22), altimeter (7, figure 1-22), turn and bank indicator (56, figure 1-22), and rate-of-climb indicator (54, figure 1-22). Of the above instruments, the gyro horizon and directional gyro are operated on three phase ac power from the instrument inverters, the airspeed indicator by static and dynamic atmospheric pressure, and the rate of climb and altimeter by static atmospheric pressure. The turn and bank indicator operates from the 28-volt dc supply. To the right of the engine instruments and directly in front of the copilot are an airspeed indicator (13, figure 1-22), remote indicating compass (15, figure 1-22), rate gyro control (turn-and-bank) (16, figure 1-22), vertical gyro (17, figure 1-22), altimeter (14, figure 1-22), and rate-of-climb indicator (18, figure 1-22). The rate gyro control, and vertical gyro control are components of the automatic pilot, and the automatic pilot inverter switch must be ON before these instruments are operative. Either the main or spare single phase inverter and the automatic pilot inverter must be ON before the remote indicating compasses on the radio magnetic indicators become operative. An ID/250 course indicator (39, figure 1-22) is located in the top center of the instrument panel on airplanes having dual radio compass installation. Airplanes equipped with only one radio compass have a remote indicating compass installed in this space. Above the instrument panel, in the windshield vee, a magnetic compass is installed on some airplanes. On the airplanes in which Radar Navigation and Search equipment (AN/APS-42) is installed, the magnetic compass is mounted immediately above the automatic pilot emergency disconnect handle at the copilot's station.

Note

Due to the proximity of the pilot's APS-42 indicator, the magnetic compass may be affected by the magnetic field within the indicator. Consequently, errors in magnetic compass headings may result when the pilot's indicator is installed. When the indicator is removed and stowed under the navigator's table, magnetic compass readings are reliable.

The magnetic compass light is controlled by a rheostat (11, figure 1-6) on the overhead panel.

INSTRUMENT



1. WINDSHIELD WIPER RHEOSTAT
2. ALTITUDE INDICATOR LIGHT-BELOW (EARLY AIRPLANES)
3. ALTITUDE INDICATOR LIGHT-PRE-SET (EARLY AIRPLANES)
4. ALTITUDE INDICATOR LIGHT-ABOVE (EARLY AIRPLANES)
5. RADIO ALTITUDE INDICATOR (EARLY AIRPLANES)
6. COURSE INDICATOR
7. ALTIMETER
8. AIRSPEED INDICATOR
9. DIRECTIONAL GYRO
10. MANIFOLD PRESSURE PURGE VALVES
11. GYRO HORIZON INDICATOR
12. HOT PRIME TEMPERATURE GAGES
13. AIRSPEED INDICATOR
14. ALTIMETER
15. REMOTE INDICATING COMPASS
16. RATE GYRO (TURN-AND-BANK)
16A. FIRE DETECTOR MASTER WARNING LIGHT
17. VERTICAL GYRO
18. RATE-OF-CLIMB INDICATOR
19. MARKER BEACON LIGHT
20. PROPELLER DE-ICING AMMETER (LEFT)
21. PROPELLER DE-ICING AMMETER (RIGHT)
22. TAIL ANTI-ICING INDICATOR
23. WING ANTI-ICING INDICATOR
24. DOOR WARNING LIGHT
25. MAIN INVERTER FAILURE WARNING LIGHTS
26. AUTOMATIC PILOT INVERTER WARNING LIGHT
27. INSTRUMENT INVERTER EMERGENCY POWER SWITCH
28. INSTRUMENT INVERTER FAILURE WARNING LIGHTS
29. AC VOLTMETER AND VOLTAGE SELECTOR SWITCH
30. DC VOLTMETER AND VOLTAGE SELECTOR SWITCH
31. RADIO MAGNETIC INDICATOR
32. RIGHT ENGINE GENERATOR AMMETER AND WARNING LIGHT
33. APP GENERATOR AMMETER AND WARNING LIGHT
34. LEFT ENGINE GENERATOR AMMETER AND WARNING LIGHT
35. WING FLAP POSITION INDICATOR
36. LANDING GEAR EMERGENCY SWITCH
37. LANDING GEAR INDICATOR LIGHTS AND WARNING HORN DISCONNECT BUTTON
38. LANDING GEAR NORMAL OPERATION SWITCH
39. REMOTE INDICATING COMPASS
40. CYLINDER HEAD TEMPERATURE INDICATORS
41. CARBURETOR AIR TEMPERATURE INDICATORS
42. OIL PRESSURE INDICATORS
43. OIL TEMPERATURE INDICATORS
44. OIL QUANTITY INDICATORS
45. FUEL QUANTITY INDICATORS AND TEST BUTTONS
45A. WATER INJECTION QUANTITY GAGE
45B. WATER INJECTION PRESSURE GAGES
46. FUEL PRESSURE INDICATORS
47. FUEL FLOWMETERS
48. TORQUEMETERS
49. TACHOMETERS
50. MANIFOLD PRESSURE GAGES
51. HYDRAULIC HIGH PRESSURE GAGE
52. HYDRAULIC LOW PRESSURE GAGE
53. CLOCK
54. RATE-OF-CLIMB INDICATOR
55. RADIO MAGNETIC INDICATOR
56 TURN AND BANK INDICATOR
57. ALTITUDE LIMIT SWITCH (EARLY AIRPLANES)

Figure 1–22.

PANEL

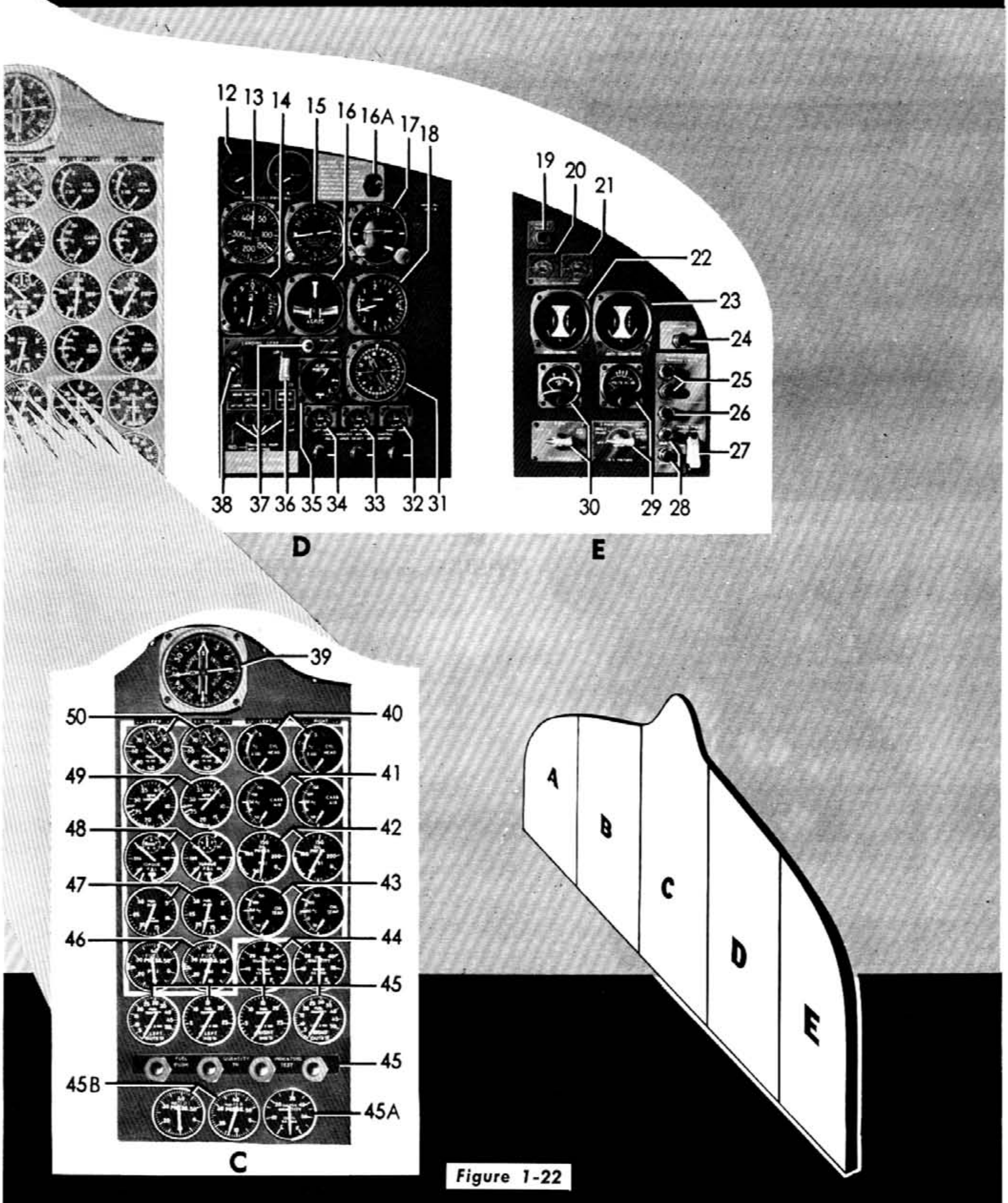
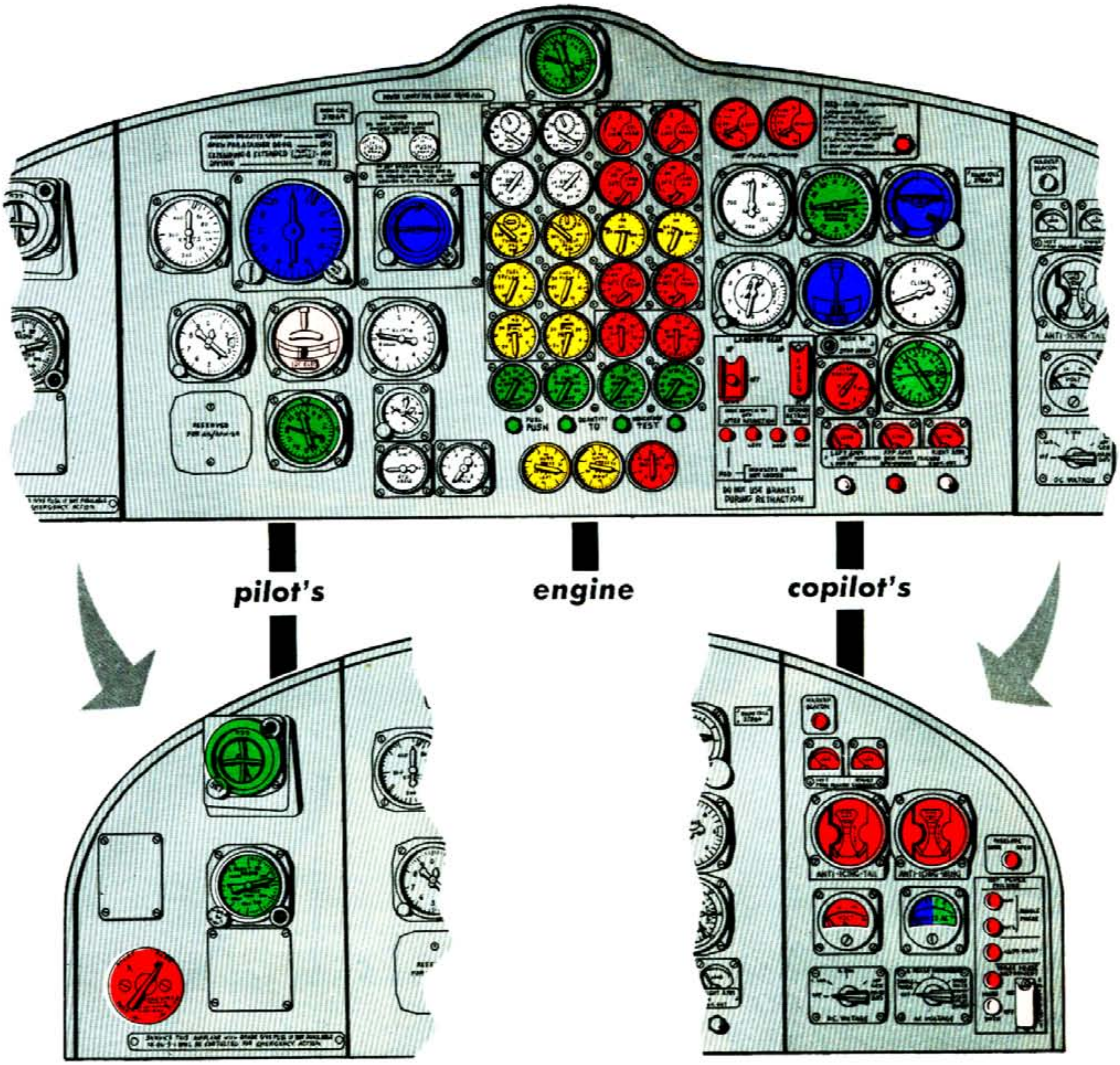


Figure 1-22

INSTRUMENT POWER *Diagram*



KEY

Red	28 VOLTS DC
White	28 VOLTS DC (EMERG.)
Yellow	26 VOLTS AC
Green	115 VOLTS AC (1 PHASE)
Blue	115 VOLTS AC (3 PHASE)

Figure 1-23

THERMOMETERS.

Free air temperature gages, calibrated in degrees Centigrade, are mounted on the windows in the right side of the crew compartment.

IGNITION ANALYZER EQUIPMENT.

Provisions have been made for the installation of a portable ignition analyzer for use either during flight or ground operation. All necessary equipment except the analyzer itself has been installed, the connecting cables and leads being stored in the lead storage locker located at the navigator's station. Basically, this equipment provides a means of analyzing an inoperative or malfunctioning ignition system and determining the cause of the system's abnormal operation.

EMERGENCY EQUIPMENT.

FIRE DETECTOR SYSTEM.

The fire detector system is incorporated in the airplane to provide warning of excessive temperature conditions where controlled combustion is employed in normal operation. Existence of any fire or excessive rise in temperature in these controlled combustion units is indicated by a master fire detector warning light on the instrument panel. Exact location of the fire is further indicated by four area warning lights on the overhead panel. Thermocouple detector units are installed at various points in each engine nacelle, the heater compartment, and APP installation. In the event of fire, these units will signal the pilot in one second. The warning light system also incorporates a push-to-test button located on the overhead panel for complete testing of the detector units and the warning lights. Power for the operation of the fire detector system is obtained from the 28-volt dc electrical system.

FIRE DETECTOR MASTER WARNING LIGHT.

The fire detector master warning light (16A, figure 1-22) on the instrument panel will glow when the detector thermocouples detect excessive rise in temperature in either engine, the heater compartment, or APP.

FIRE DETECTOR AREA WARNING LIGHTS.

Four indicator lights (23, 27, 28, 30, figure 1-6) on the overhead panel indicate the immediate area in which the fire is located. An area warning light is provided for each engine, the heater compartment, and APP.

FIRE DETECTOR TEST BUTTON.

When depressed, the fire detector test button (26, figure 1-6) on the overhead panel energizes a separate 28-volt dc test circuit providing a check not only of the warning lights but the complete fire detector system.

Note

When testing the fire detector system, allow at least 15 seconds for indicator lights to glow. This permits sufficient time for complete energization of the test system.

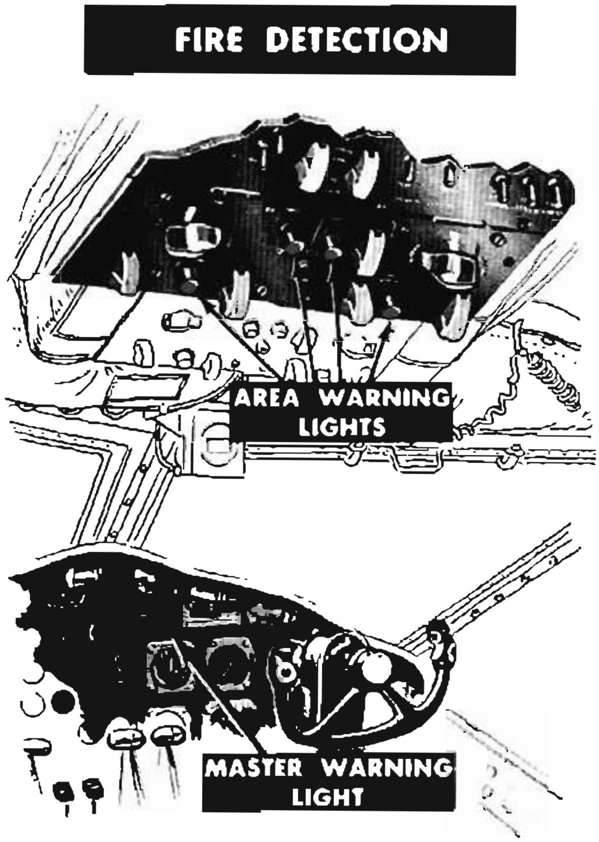


Figure 1-24

FIRE DETECTOR TEST SWITCH.

On later airplanes, the fire detector system is split into two divisions. One section of the system comprises half the thermocouples located on each engine baffle and in the accessory section; the other section includes not only the remaining engine and engine accessory compartment thermocouples but also those located in the heater compartment and APP installation. A three-position switch, located on the overhead panel, is utilized to test each of the two divisions of the system. When the switch is turned to the L. ENG.-R. ENG. position, one half of the detector system in both nacelles is tested; in the L. ENG.-R. ENG.-HTR-APP position, the remaining half of the engine and engine accessories detector circuits are tested as are the detector circuits in the heater compartment and APP. The center position of the switch is OFF.

Note

When testing the fire detector system, allow at least 15 seconds for indicator lights to glow. This permits sufficient time for complete energization of the test system.

FIRE EXTINGUISHER SYSTEM.

When a fire is indicated by the master warning light and area warning light, the pilot or copilot operates the appropriate fire extinguisher control switch. This electrically energizes the 28-volt dc cylinder release valve allowing extinguishing agent to be released from the cylinder into the distribution system. Distribution of the extinguishing agent in each engine is accomplished by means of a fire extinguisher ring with nozzles; a stainless steel tube routed across the entire heater compartment, likewise, performs the same distributive function. Hand fire extinguishers (12, figure 3-3) should be employed to combat a fire in the APP.

ENGINE FIRE EXTINGUISHER SWITCHES.

WARNING

Repeated or prolonged exposure to high concentrations of bromochloromethane (CB) or decomposition products should be avoided. CB is a narcotic agent of moderate intensity but of prolonged duration. It is considered to be less toxic than carbon tetrachloride, methyl bromide, or the usual products of combustion. In other words it is safer to use than previous fire extinguishing agents. However, normal precautions should be taken including the use of oxygen when available.

Two engine fire extinguisher switches (24, 29, figure 1-6), one for each engine installation, are located on the overhead panel. Operation of either switch to the ON position directs current (28-volt dc) to the cylinder discharge cartridge which consists of a slug, a charge of explosive powder, and a low resistance fuse wire. The slug, propelled by the explosion of the powder, breaks a zinc seal permitting the bromochloromethane to be distributed throughout the system. After either engine fire extinguisher system is actuated, the full charge of the respective system is depleted necessitating replacement of the extinguisher bottle before that system is again operative. Excessive heat causes a safety plug on the cylinder to release the extinguishing agent into the nacelle area. A pressure gage on the bottle itself indicates the condition of the charge. The pressure as indicated by the gage is subject to marked variations with changes in temperature.

HEATER FIRE EXTINGUISHER SWITCH.

Note

In order to prevent fumes from entering the crew compartment during operation of the heater fire extinguisher system, the cockpit air switch should be turned OFF.

Two CO_2 cylinders are provided to extinguish fires in the heater compartment. Each 28-volt dc cylinder release valve is actuated by means of a solenoid located on the upper end of the cylinder. When the two-position heater fire extinguisher switch (25, figure 1-6) on the overhead panel is placed in the ON position, the solenoids are energized, releasing the extinguishing agent of both CO_2 cylinders into stainless steel supply lines routed throughout the heater compartment. Cylinder charge is checked by weighing according to the weight data appearing on the cylinder placard.

HEATER FIRE EXTINGUISHER DISCHARGE INDICATORS.

Disc type discharge indicators for the two CO_2 cylinders are located on the left rear side of the fuselage and give a visual indication of the condition of the cylinders. Ejection of the red overheat indicator disc denotes cylinder dissipation due to expansion or leakage, while ejection of the yellow disc indicates the cylinder has been discharged by operation of the system.

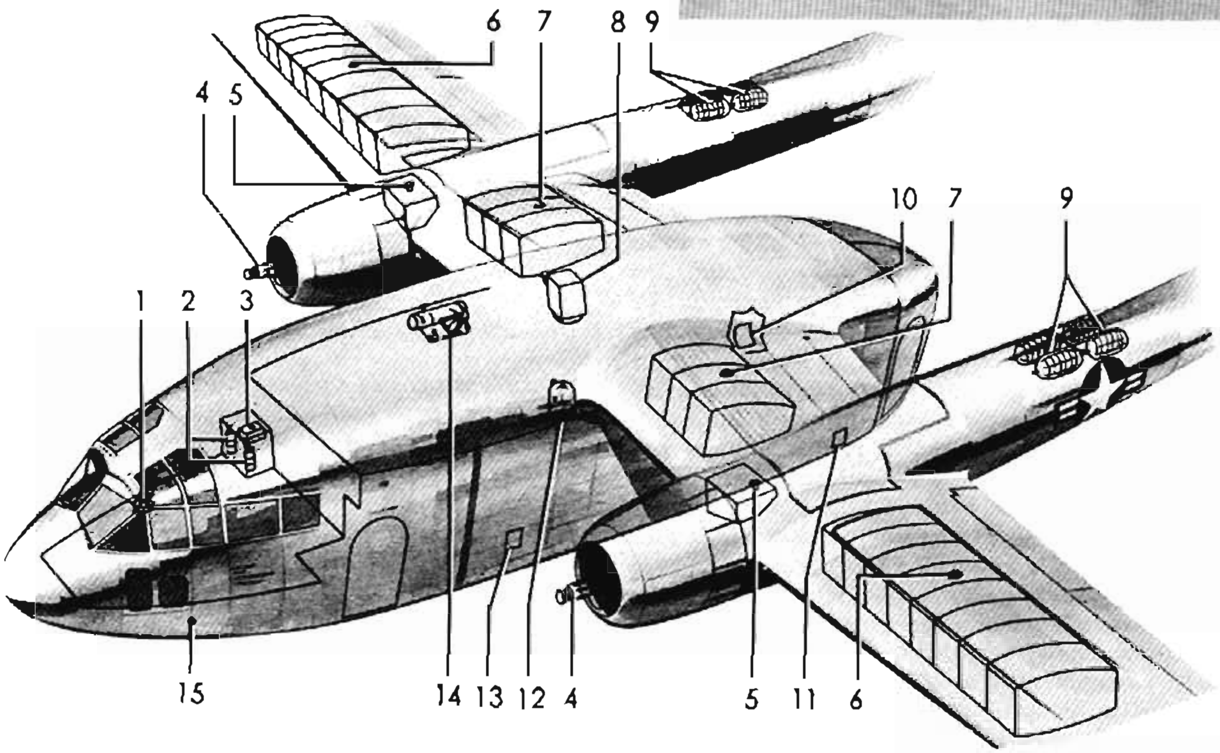
HAND FIRE EXTINGUISHERS.

A bromochloromethane (CB) hand fire extinguisher (12, figure 3-3) is installed on the forward part of the auxiliary floor, another on the left rear side of the cargo compartment, and a third inside the forward entrance door accessible from the ground. A pressure gage mounted on each extinguisher indicates the pressure charge existing within the extinguisher.

WARNING

Repeated or prolonged exposure to high concentrations of bromochloromethane (CB) or decomposition products should be avoided. CB is a narcotic agent of moderate intensity but of prolonged duration. It is considered to be less toxic than carbon tetrachloride, methyl bromide, or the usual products of combustion. In other words it is safer to use than previous fire extinguishing agents. However, normal precautions should be taken including the use of oxygen when available.

SERVICING DIAGRAM



1	EXTERNAL POWER RECEPTACLE			
2	DRINKING WATER CONTAINERS			
3	LAVATORY WATER TANK			
4	PROPELLER OIL	MIL-O-5606		
5	OIL TANKS	MIL-L-6082	1100	
6	FUEL TANKS (OUTBOARD)	MIL-F-5572	115/145	
7	FUEL TANKS (INBOARD)	MIL-F-5572	GRADE 115/145 GRADE 100/130 (ALTERNATE)	
8	WATER INJECTION SUPPLY TANK	ALCOHOL SPECIFICATION O-M-232 GRADE A	ALCOHOL 50%	*WATER 50%
		ALTERNATE SPECIFICATION MIL-A-6091	50%	50%

NOTE . . . Water used should be distilled or chemically demineralized, with 2/3% Oil-Emulsion SPEC MIL-C-4339 added.

9	OXYGEN CYLINDERS	AN-O-1	A-DRIED
10	BATTERY ACCESS		
11	OXYGEN FILLER	AN-O-1	A-DRIED
12	HYDRAULIC RESERVOIR	MIL-O-5606	
13	HYDRAULIC POWER RECEPTACLE	MIL-O-5606	
14	APP OIL SUPPLY	MIL-L-8383	1065
15	AIR BRAKE FILLER		

Figure 1-25

PYROTECHNIC EQUIPMENT.

A pyrotechnic pistol and flares (4, figure 3-3) are stowed in a canvas container on the left side of the shelf above the radio rack just aft of the navigator's station. A mount for holding the pistol in the firing position is located in the ceiling at the right aft end of the crew compartment in such a manner that the pistol may be fired through the top of the fuselage. A flare chute is provided in the floor at the navigator's station.

BAIL-OUT SIGNAL LIGHTS AND ALARM BELLS.

Red and green bail-out signal lights are installed just forward of the cargo jump doors, the red being used as a warning, and the green being used as a signal for the troops to begin the jump. A duplicate set of lights, which may be dimmed by rotating the light cover, is provided in the crew compartment adjacent to the control switches (3, figure 1-17), (1, figure 1-18) located on the pilot's and copilot's jump signal panels. The control switches are so designed that the red light must be turned ON before the green light may be illuminated. An emergency bell controlled by an OFF-ON switch on each of the jump signal panels is located near the cockpit entrance hatch in the forward part of the cargo compartment and an additional bell is installed in the cargo doors. The bells are so placed that they are audible to the crew members and all personnel in the cargo compartment. Actuation of the light system and/or bell system may be accomplished with the battery switch in the OFF position. Additional lights are installed in each boom of the airplane facing the troop jump doors.

DOOR WARNING LIGHT SYSTEM.

A door warning red indicator light (24, figure 1-22) on the instrument panel will glow when the front entrance door is open.

ESCAPE HATCHES.

A panel around the navigator's astrodome may be jettisoned by a quick-release handle (1, figure 3-3) to provide an escape exit for crew members on the ground or during ditching.

BAIL-OUT CHUTE.

A bail-out chute located in the crew compartment floor at the flight mechanic's station is provided to permit a safe exit when bail-out is necessary. A handle, recessed in the top door which forms a section of the crew compartment floor, is used to operate both the inner and the outer door. Raising the top door, through a system of push rods, releases latches in the forward end of the outer door, and allows it to fall free of the airplane. The top door of the bail-out chute should be safety-wired in such a manner that the door can be raised approximately 1 inch from the jamb. This will prevent the releasing of the outer door when checking the top door for free and unrestricted movement.

DITCHING HATCHES.

A hatch is provided on each upper side of the cargo compartment aft section for egress of personnel when it becomes necessary to ditch the plane. The ditching hatches are provided with a lever-type latching mechanism operable from both inside and outside the airplane (8, figure 3-3). The rungs of the ladders to the hatches are painted yellow.

EMERGENCY AXE.

An emergency axe (15, figure 3-3) is stowed at the crew compartment entrance to the right of the flight mechanic's seat.

FIRST AID KITS.

Two first aid kits (5, figure 3-3) are installed in the crew compartment. Four first aid kits are installed in the cargo compartment and mounting provisions are provided for the installation of four additional kits.

FLASHLIGHTS.

Four flashlights (7, figure 3-3) are installed at the aft end of the cargo compartment: one at each paratroop door and one at each overhead escape hatch. A flashlight is also installed just forward of the front entrance door. A flashlight in the crew compartment is mounted at the navigator's station.

LIFE RAFTS.

A seven-man life raft is stowed in the life raft compartment, located in the upper aft section of the cargo compartment in such a manner that it may be automatically released and inflated by either of two interior release handles. One release handle (6, figure 3-3) is located on the upper bulkhead at the aft end of the crew compartment, and the other release handle (13, figure 3-3) is installed in the cargo compartment just aft of the forward entrance door. When either of these handles is pulled the life raft compartment access door will be released and a valve installed in the compartment will open to inflate the raft from a CO_2 cylinder. If, for any reason it is impossible to pull either of these handles before the airplane is abandoned, the life raft may be released by means of an external handle (9, figure 3-3) in the top of the fuselage, just aft of the life raft compartment access door. However, when the external release handle is utilized, the release valve on the CO_2 cylinder must be tripped by hand. When the airplane is used for paratroop operations, extra life rafts may be carried in the cargo compartment. The regular tie-down fittings are used for securing these auxiliary life rafts.

QUICK-RELEASE DOOR HANDLES.

The front entrance door and both paratroop doors are equipped with quick-release hinges. In case of emergency, the paratroop doors may be quickly released by disengaging the safety pin and pulling up on release handle (11, figure 3-3) thus extracting the hinge pins. The front entrance door may be opened from the inside of the airplane by pulling down on the release handle (14, figure 3-3).

SEATS.

The pilot's and copilot's seats may be moved for vertical, lateral, forward, and rear adjustment, and adapted for reclining. The manual control (8, figure 1-18), at the left of the seat, controls forward, aft, and lateral adjustment. To the right of the seat is the vertical adjustment lever (13, figure 1-17) and the reclining adjustment lever (14, figure 1-17).

OPERATIONAL EQUIPMENT.

The airplane is equipped with a complete oxygen system, communications equipment, an auxiliary power plant, a heating, ventilating and anti-icing system, aerial delivery, and equipment drop systems. Refer to Section IV for a detailed description and full operating procedures.