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Flight Handbook

NAVY MODEL

HR2S-1

HELICOPTERS

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FOREWORD

IMPORTANT

In order that you will gain the maximum benefits from this handbook, it is important that you read this page carefully.

The function of this handbook is to describe the helicopter, its equipment, and its operation so that a flight crew will have sufficient information to accomplish a safe flight efficiently. This handbook is divided into nine sections and an appendix, each of which contains a particular phase of description or operating instructions. A brief description of the subject matter contained in each section is as follows:

SECTION I, DESCRIPTION.

The function of this section is to describe the helicopter, its systems, and controls which are essential to flight. Included also in this section is all of the emergency equipment that is not part of an auxiliary system.

SECTION II, NORMAL PROCEDURES.

This section contains the steps of procedure to be accomplished from the time the helicopter is approached by a flight crew until it is left parked on the ramp after accomplishing one complete non-tactical flight under normal conditions. The information in this section has been gathered from experimental data, the engine manufacturer, and flight tests.

SECTION III, EMERGENCY PROCEDURES.

The function of this section is to describe the procedure to be followed in meeting emergencies that could reasonably be expected to be encountered. The information in this section has been gathered from experimental data and has not been thoroughly flight tested.

SECTION IV, DESCRIPTION AND OPERATION OF AUXILIARY EQUIPMENT.

This section includes the description, normal operation, and emergency operation of all equipment not directly contributing to flight, but which enables the helicopter to perform certain specialized functions.

SECTION V, OPERATING LIMITATIONS.

This section includes instrument range markings and other limitations that have been established at the time of writing of this handbook.

SECTION VI, FLIGHT CHARACTERISTICS.

This section discusses the unique flight characteristics of this helicopter and its controls during normal flight.

SECTION VII, SYSTEMS OPERATION.

A further discussion of the operation of the engines, the hydro-mechanical clutches, the fuel system, and the throttles is contained in this section.

SECTION VIII, CREW DUTIES.

This section covers the duties of the pilot, copilot, and a crew member.

SECTION IX, ALL-WEATHER OPERATION.

This section provides information relative to operation under various weather conditions.

APPENDIX I, OPERATING DATA.

This appendix contains operating data charts required for pre-flight and in-flight planning. A flight operation instruction chart has been included for single-engine operation.

Figure 1—1. HR2S-1 Helicopter

SECTION I
DESCRIPTION

THE HELICOPTER.

The model HR2S-1 helicopter, manufactured by Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut, is a twin-engine, all-metal helicopter designed for assault transport and cargo operations. The rotor system is composed of a five-bladed single main rotor and a four-bladed anti-torque tail rotor. The engines are mounted in nacelles at the tips of a short high wing. Fuel is carried in wing and nacelle tanks and the retractable main landing gear is mounted in the nacelles, leaving the main section of the fuselage clear for loading and unloading troops or cargo. The lower portion of the nose of the fuselage consists of "clam-shell" type cargo loading doors and a cargo loading ramp. The pilot's compartment is located in the upper portion of the nose of the helicopter and contains seats, controls, and instruments for one pilot and one copilot. The large windshield and the side and top windows in the pilot's compartment and the windows in the clam-shell doors provide good visibility from the pilot's compartment. Electronic equipment is installed behind the pilot's and copilot's seats and in the aft compartment.

Aft of the pilot's compartment and over the wing center section is the main transmission compartment housing the main gear box. The engines are installed in the nacelles with the engine shafts pointing aft and inboard within the wing. Hydro-mechanical clutches connect the engine shafts to shafts extending from the base of the main gear box. Power is transmitted from the engines to both the main and tail rotor through three gear boxes and connecting shafting. The main rotor assembly, with five fully-articulated folding blades, is mounted on top of the main rotor drive shaft. The two-cell fuel tank for each engine is located within each wing and in the aft portion of each nacelle. A jettisonable auxiliary fuel tank may be mounted externally below each wing to increase the range of the helicopter. The cabin of the helicopter is entered through the cargo door on the right side of the fuselage, the passenger door within the cargo door, or through the clam-shell nose doors. The cabin is capable of carrying 36 fully equipped troops, 24 litters, or equipment up to a 105 mm howitzer and trailer. A cargo hatch is located in the center of the cabin door. A cargo sling capable of carrying 1000 pounds is installed under the fuselage; and a cargo hoist with a maximum capacity of 2000 pounds, mounted on a rail in the cabin, may be used for loading at the nose door, side cargo door, or floor hatch. Behind the cabin aft bulkhead is the aft fuselage compartment. The tail rotor pylon, which supports the tail rotor assembly and also functions as a fin, is mounted at the aft end of the fuselage and is hinged at its right-hand attachment point to permit the pylon to be folded forward along the rear portion of the fuselage. A dorsal fin is installed on top of the aft fuselage section. The tail rotor assembly, consisting of four fully-articulated blades and the tail rotor gear box, is installed on the upper end of the tail rotor pylon. The intermediate gear box is mounted at the lower end of the tail rotor pylon. A full-castering, self-centering tail wheel is mounted at the aft end of the fuselage. A two-position stabilizer, fixed in the forward flight position and free-swinging in the hover position, is installed in the aft fuselage section.

1. Tail Rotor Assembly
2. Intermediate Gear Box Cooling Air Inlet
3. Pylon Fold Jury Strut
4. Static Ports
5. Heating and Ventilating Air Intake (Left Side)
6. Service Walkway
7. Access Footwells (Left Side)
8. Engine Preheat Duct Connections
9. Main Gear Box Oil Cooling Air Outlet
10. Main Rotor Assembly
11. Main Gear Box Oil Cooling Air Intake
12. Rotor Clutch and Engine Cooling Fan
13. Engine Access Door
14. Carburetor Cold Air Intake
15. Free-Air Thermometer
16. Windshield Wiper
17. Nose Doors
18. Gun Ports
19. Pilot's Compartment Access Footwells
20. Pitot Tubes
21. Main Landing Gear
22. Mooring Fittings
23. Engine Cooling Air Outlets
24. Engine Exhaust Ejector Ducts
25. Walkway
26. Passenger Door and Cargo Door
27. Stabilizer
28. Stabilizer Damper Strut
29. Tail Wheel
30. Mooring Fitting
31. Pylon Hinges

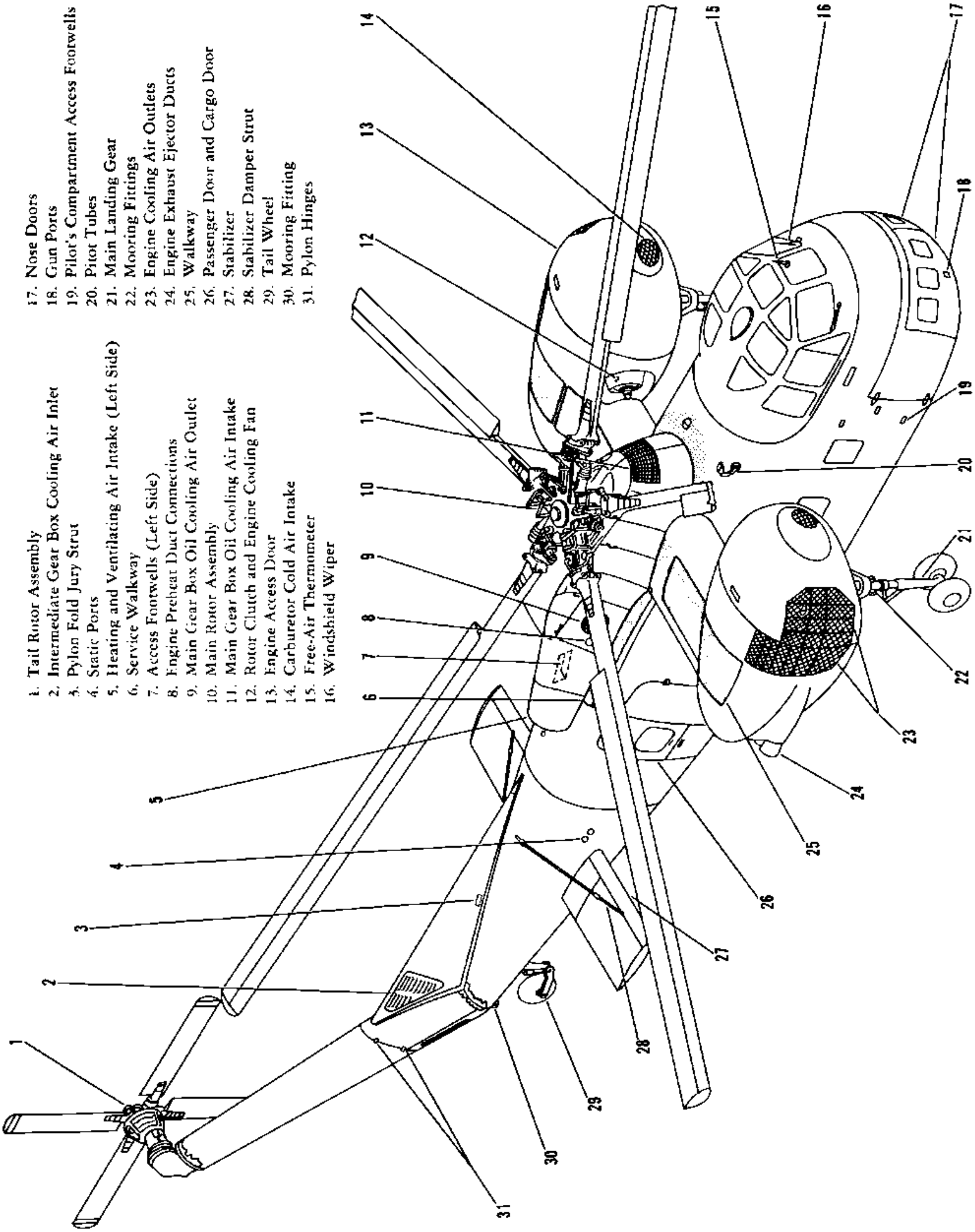


Figure 1–2. General Arrangement — Exterior

1. AN/ARC-2A
2. AN/ARC-44
3. Main Gear Box Oil Cooler
4. Main Rotor Servo Units
5. Inverters
6. Radio Equipment
7. Pilot's Compartment
8. Pilot's Compartment Ladder
9. Cargo Release Handle Access Door
10. Battery
11. Auxiliary Fuel Tank Jettison Handle (Both Sides)
12. External Power Receptacle
13. Generator (Both Engines)
14. Engine
15. Cabin
16. Auxiliary Power Unit
17. AN/PRC-8, -9, or -10
18. Aft Cabin Bulkhead
19. Aft Compartment
20. Aft Cabin Bulkhead Door

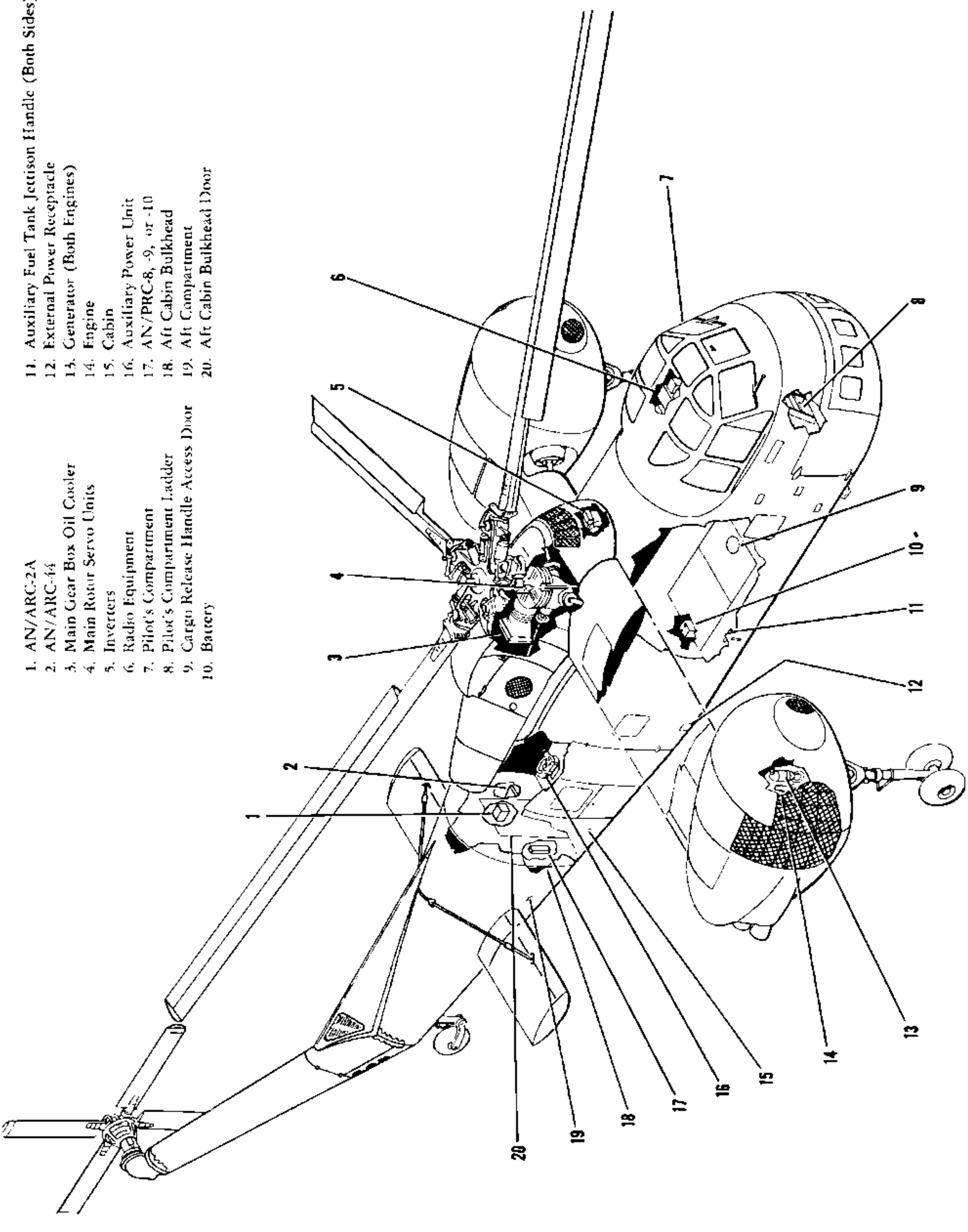


Figure 1—3. General Arrangement — Interior

DIMENSIONS.

Length: maximum (with both rotors at extreme positions) .... 88′ 3″

Length: minimum (main rotor blades and tail pylon folded) .... 55′ 8″

Width: blades and pylon folded .... 27′ 4″
Height: maximum — to top of rotor blade (tail pylon unfolded) .... 22′ 1″

Height: minimum — to top of main rotor (tail pylon folded) (with main landing gear stowage valve actuated, 6 inches less) .... 17′ 1″

Main rotor disc diameter .... 72′ 0″

Tail rotor disc diameter .... 15′ 0″

Main rotor ground line clearance: minimum — forward sector .... 11′ 6″

Tail rotor ground line clearance .... 7′ 0″

Tread of main wheels .... 19′ 9″

ENGINES.

The helicopter is powered by two Pratt & Whitney, R-2800-54, 18-cylinder, twin-row radial engines (14, figure 1—3) equipped with single-stage, single-speed superchargers. Each engine is located in a nacelle at the outboard ends of the wing. The engines are mounted with the drive shafts slanting inboard at an angle of approximately 80 degrees and pointing upward at an angle of approximately 12½ degrees. A hydromechanical rotor clutch is splined to each engine drive shaft. Drive shafts connect the clutches to the main gear box. Pressure-type, downdraft carburetors are mounted on the top of the intermediate rear section of the engines. Carburetor cold air intakes (14, figure 1—2) are located in the nose of each nacelle. Cooling air is drawn in through openings in the inboard sides of the nacelles. Rotor clutch and engine cooling fans (12, figure 1—2) driven by the engines, force the cooling air over the engines and out through the screened engine cooling air outlets (23, figure 1—2) on the outboard side of the nacelles. Engine oil coolers are located outboard of the engines. Cooling air enters a duct at the bottom of each nacelle and passes upward through the oil cooler where it is accelerated and directed aft by the engine exhaust ejector ducts (24, figure 1—2). The ejector ducts also provide exhaust flame dampening. Access to all parts of the engine is provided by hinged engine access doors (13, figure 1—2) comprising the nose of each nacelle.

THROTTLES.

There are two types of throttle controls in the pilot's compartment: the control quadrant throttle and the collective pitch control twist-grip throttle. The control quadrant throttles are used for ground operation and for equalizing manifold pressures before take-off and during flight. The twist-grip throttles are used during flight and automatically vary the power settings of both engines simultaneously with changes in collective pitch. For a more detailed description of the operation of both throttles, refer to **QUADRANT THROTTLE AND TWIST-GRIP THROTTLE OPERATION,** Section VII.

CONTROL QUADRANT THROTTLES. Two throttles (figure 1—9), one controlling each engine, are located on the engine control quadrant which is mounted on the forward end of the overhead switch panel in the center of the pilot's compartment. Two throttle positions, OPEN and CLOSED, with graduations of 0 to 90 degrees between open and closed, are marked on the quadrant. The control quadrant throttles are used for individual engine starting, ground operation, and to synchronize the engines. Normally, during flight, the control quadrant throttles are not used except for minor adjustments to manifold pressure for engine synchronization or for single-engine operation. After synchronization, the throttles of both engines are controlled simultaneously by the twist-grip throttles on the collective pitch controls. The position of the throttle levers on the quadrant is a direct indication of the amount of throttle being used. The quadrant throttles are connected directly to the carburetors by control cables. Each throttle is equipped with a throttle lock, controlled

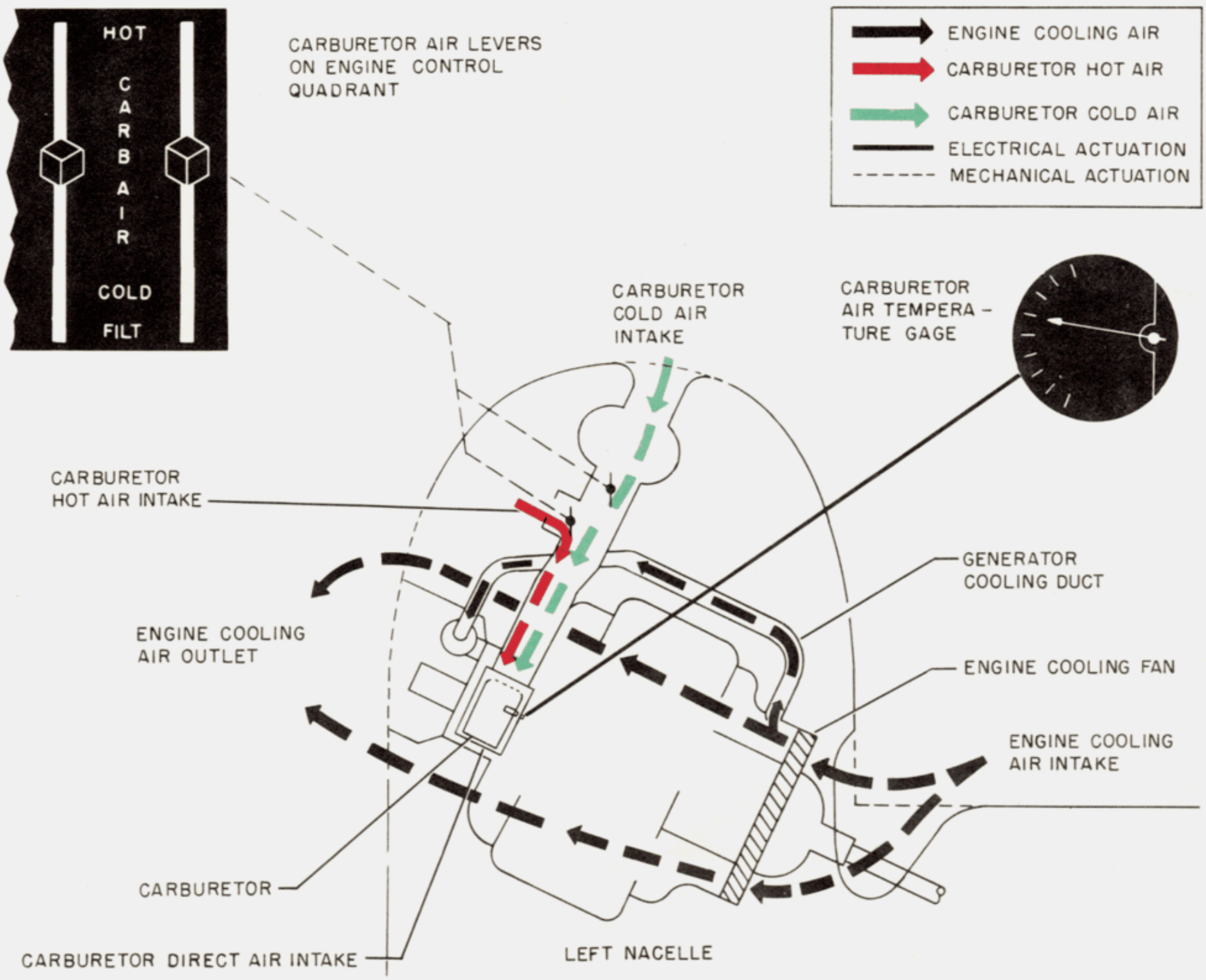
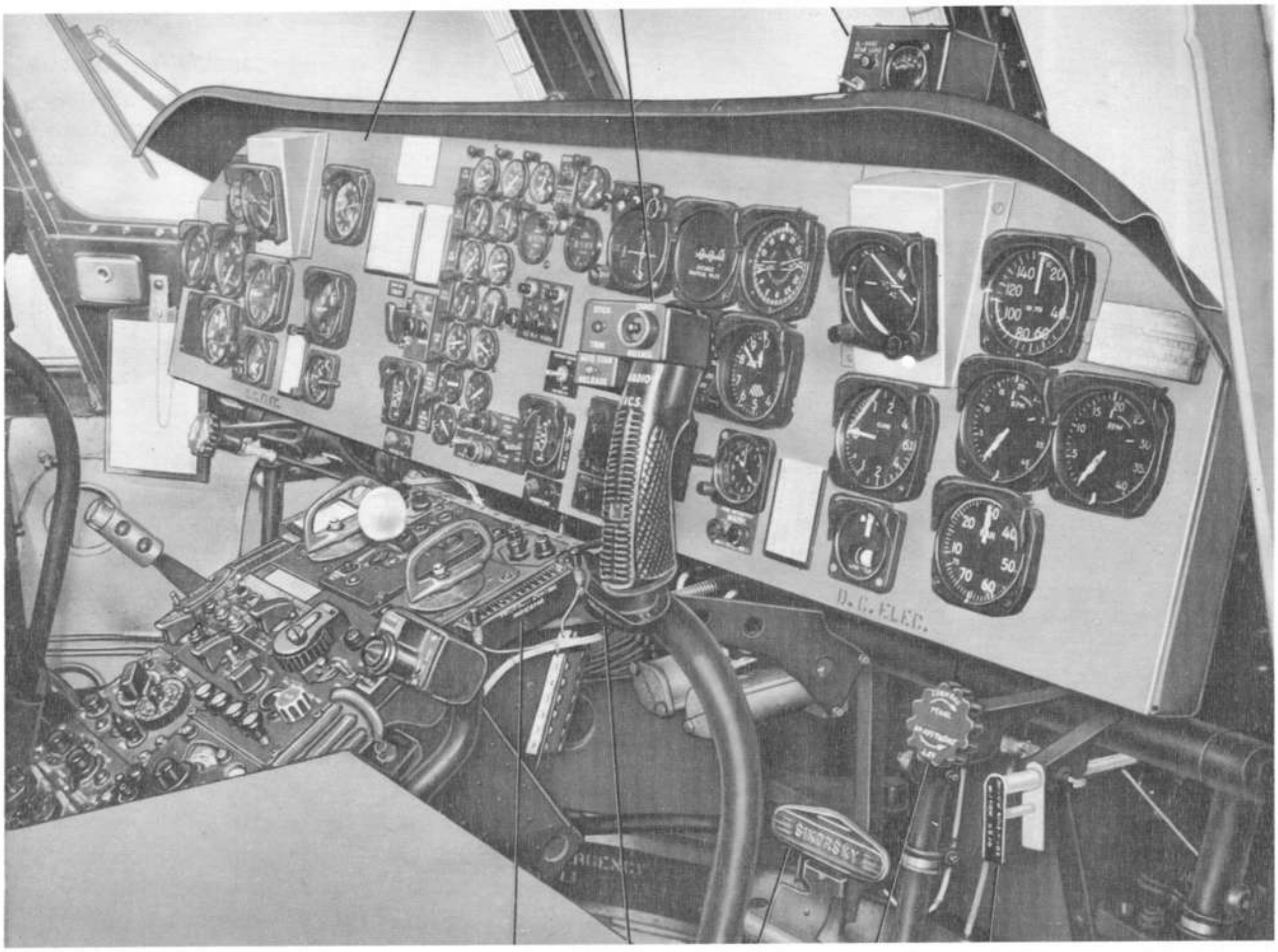


Figure 1–4. Engine Cooling and Carburetor Air Systems

by a lock lever mounted on the throttle. The throttle lock is engaged by pressing downward on the lock lever, moving the throttle to the fully closed position, and releasing the lock lever when in the fully closed position. The throttle lock is disengaged by pressing downward on the lock lever and releasing it after the throttle has been moved out of the fully closed position. When the throttle locks are engaged, a slotted stop bar limits movement of the throttles from idle speed to 1600 rpm. The locks, when engaged, prevent overspeeding of the engine before rotor clutch engagement but provide adequate rpm for full generator output. When the throttle locks are released, unrestricted movement of the throttles is available. Throttle limit switches, installed at the top of the quadrant, prevent energizing of the starter unless the throttles are fully closed and the throttle locks engaged.

TWIST-GRIP THROTTLES. The twist-grip throttles (figure 1–26) are located on the pilot's and copilot's collective pitch controls. The twist-grip throttles control both engines simultaneously and are the throttles used for normal flight. These throttles are also synchronized with the operation of the collective pitch controls so that increasing collective pitch, while holding the twist-grips to prevent rotation, automatically increases engine power, and decreasing collective pitch decreases engine power. The twist-grip throttles may be rotated at any time for finer throttle adjustment at any position of the collective pitch control. When the grip is rotated to the left, throttle is increased and when rotated to the right, throttle is decreased. A knurled nut, located on the pilot's collective pitch control, may be rotated to apply friction to the throttles or to prevent throttle creeping. Rotation of the twist-grip throttles is facilitated by a throttle servo powered from the utility hydraulic system. The throttle servo is turned on and off



1. Instrument Panel
2. Pilot's Cyclic Control Stick
3. Deleted
4. Cyclic Control Stick Positioner
5. Pilot's Pedal Adjustment Knob
6. Wheel Brake Pedal
7. Parking Brake Handle
8. Twist-Grip Throttle Position Indicator

Figure 1–5. Pilot's Compartment – Pilot's Side

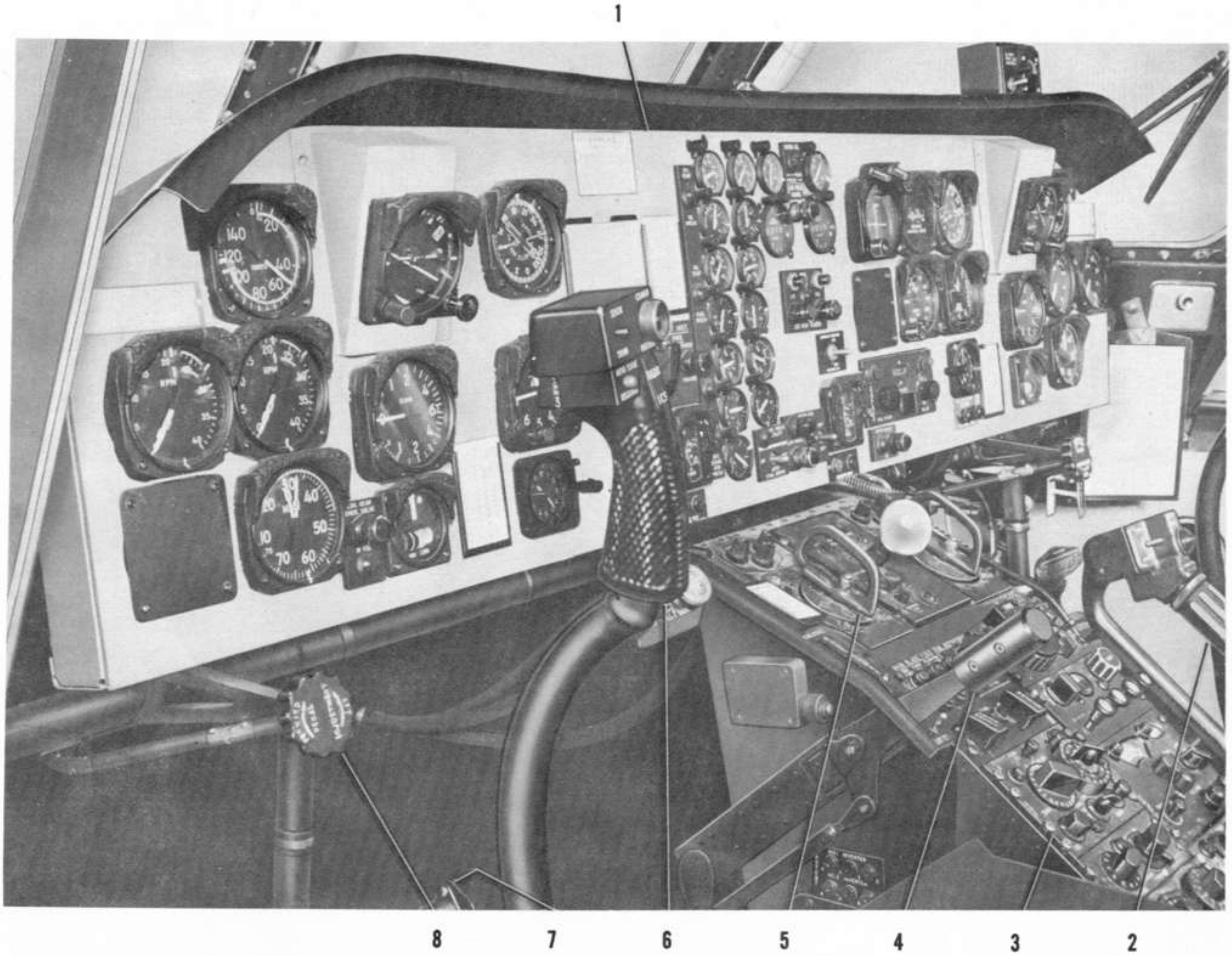
simultaneously with the tail rotor control pedal damper and the automatic stabilization system servo by actuating the switch, marked AUTO STABE SERVO, located on the instrument panel. The twist-grip throttles and the collective pitch controls are connected by the cabling and linkage to the overhead quadrant throttles through the automatic clutches in the quadrant.

TWIST-GRIP THROTTLE POSITION INDICATOR.

A twist-grip throttle position indicator (figure 1–16), located to the right of the fuel shut-off switch on the control console, shows the position of the twist-grip throttles on the collective pitch controls. The indicator consists of a nylon rod, enclosed in plastic tubing, which is connected mechanically to the throttle linkage, and a scale graduated from CLOSED through 120 degrees to OPEN. The indicator aids the pilot in maintaining the correct relationship between the quadrant throttles and the twist-grip throttle. Two lights are installed on the throttle position indicator to illuminate it at night.

MIXTURE CONTROL LEVERS.

Two mixture control levers (figure 1–9), one for each engine, are located on the section of the engine control quadrant, marked MIXTURE. The levers have positions, RICH, NORMAL, and IDLE CUT-OFF marked on the quadrant. Control cables connect the levers on the quadrant to mixture control areas on the carburetor. The RICH position is the full forward position on the quadrant. The NORMAL position is identified by a detent which prevents accidental movement of the levers into IDLE CUT-OFF. Manual leaning beyond this detent should not be attempted. The IDLE CUT-OFF position shuts off fuel flow at the carburetor. The two tachom-



1. Instrument Panel
2. Pilot's Collective Pitch Control
3. Radio Console
4. Nose Door Lock Lever
5. Control Console — Forward Section
6. Copilot's Cyclic Control Stick
7. Copilot's Tail Rotor Control Pedal
8. Copilot's Pedal Adjustment Knob

Figure 1—6. Pilot's Compartment — Copilot's Side

mixture control levers must be squeezed toward each other to pass the detent at the NORMAL position in order to move into IDLE CUT-OFF.

CARBURETOR AIR LEVERS.

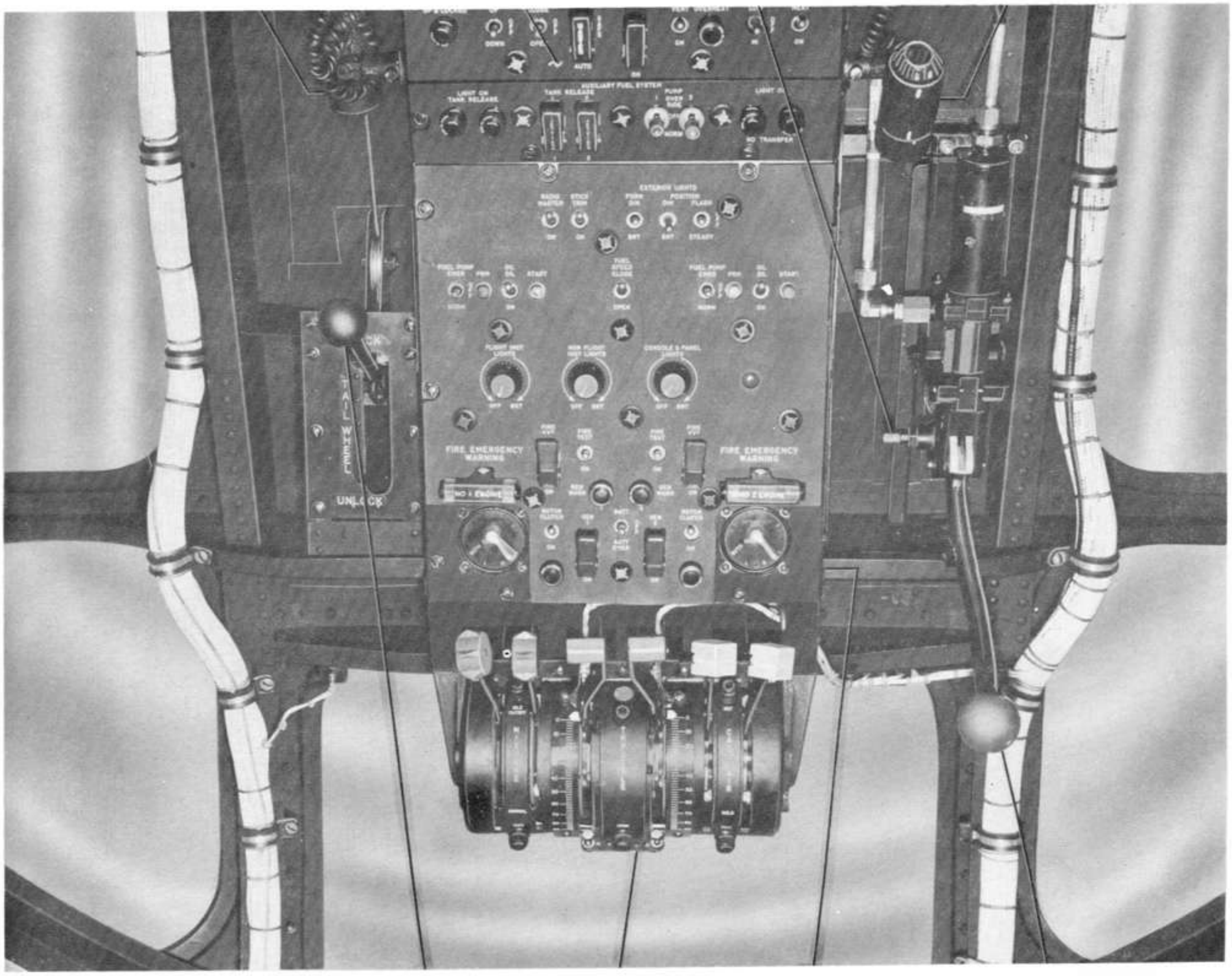
Two carburetor air levers (figure 1—9), one for each engine, are located on the section of the engine control quadrant marked CARB AIR. The levers have positions, HOT, COLD, and FILT marked on the quadrant. Each lever actuates two doors in its carburetor air intake duct by means of control cables and mechanical linkage. When a lever is in either HOT or COLD, or in any intermediate setting, the doors are adjusted to control the mixture of unfiltered cold and heated air entering the carburetor. Heated air is drawn from inside the nacelle and cold air is drawn directly through the carburetor cold air intake. The FILT position of the carburetor air levers is inoperative. Provisions are made for the installation of filters in the carburetor cold air ducts.

IGNITION SWITCHES.

Two standard-type ignition switches (figure 1—10), one for each engine, are located on either side of the overhead switch panel just aft of the engine control quadrant. The positions, OFF, R, L, and BOTH, are marked on each switch. Each ignition switch grounds the ignition vibrator and the left and right magnetos for its engine, preparatory to starting.

STARTER SWITCHES.

Two push-button switches (figure 1—10), marked START, are mounted on the overhead switch panel. Before operating the starter, the collective pitch control must be in the minimum pitch position, and the quadrant



1. Copilot's Spotlight
2. Pilot's Compartment Dome Light Panel
3. Rotor Brake Lever Lockpin
4. Pilot's Spotlight
5. Rotor Brake Lever
6. Overhead Switch Panel
7. Engine Control Quadrant
8. Tail Wheel Lock Lever

Figure 1—7. Pilot's Compartment — Overhead

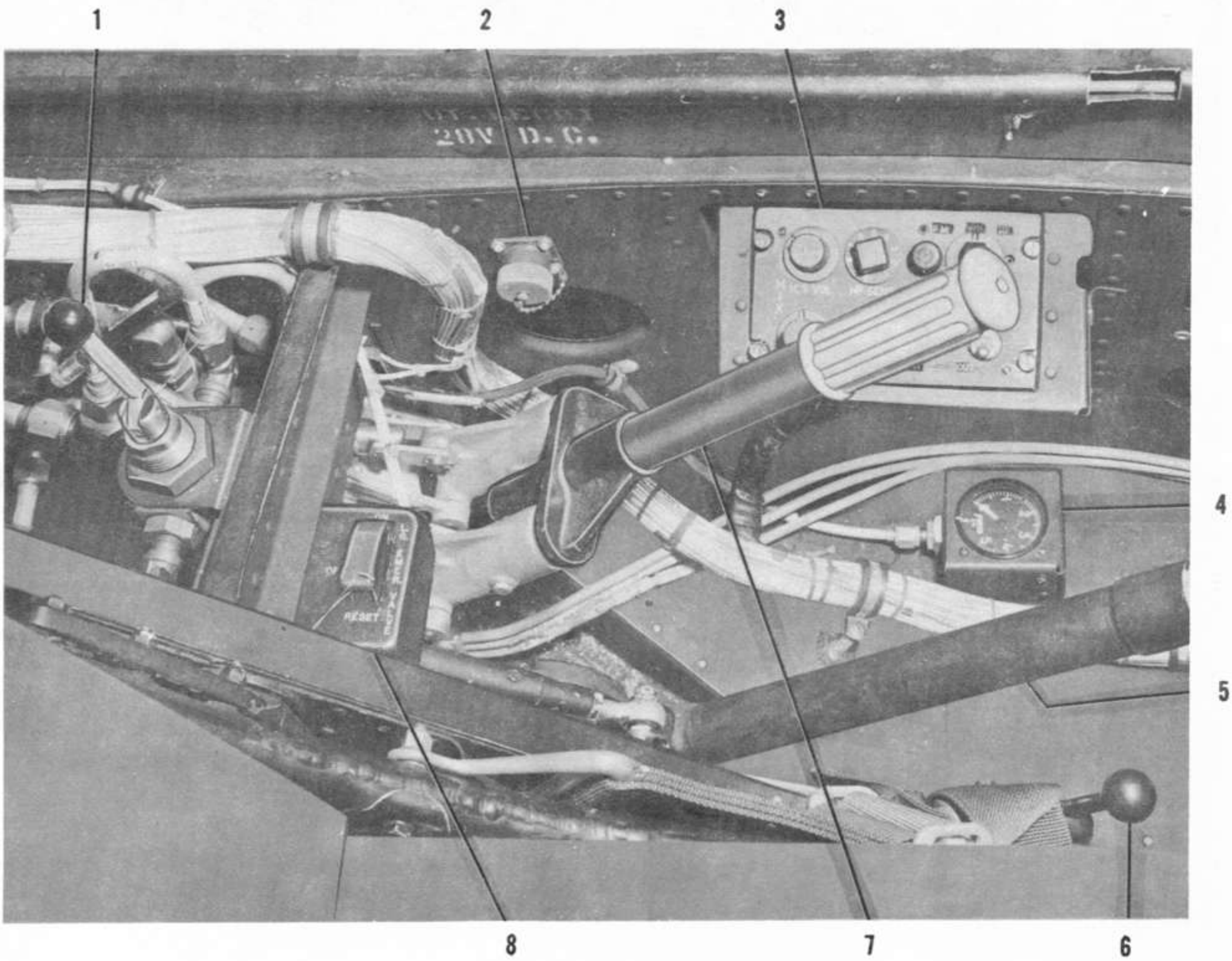
throttles in the fully closed and locked position. Limit switches on the quadrant throttles prevent the accidental engagement of the starters when the throttles are not closed and locked. When either starter button is depressed, direct current from the secondary bus actuates a relay which directs power from the primary bus to the starter and to the ignition vibrator. When the starter button is released, the circuit to the starter and to the vibrator is broken, and ignition system voltage is developed by the engine-driven magnetos.

ENGINE PRIMER SWITCHES.

Two engine primer push-button switches (figure 1—10), marked PRM, are mounted on the overhead switch panel. For engine priming, the button is actuated, as required. The primer switch utilizes direct current from the secondary bus to operate a solenoid on the carburetor which directs fuel to discharge nozzles upstream of the impeller section of the engine. Fuel pressure for priming is supplied by the fuel booster pumps. The primer circuit is protected by a circuit breaker, marked ENG START, PRM, and OIL DIL, located on the overhead circuit breaker panel.

ENGINE-ROTOR TACHOMETERS.

On BuNos 140314 and subsequent, four dual tachometers are installed on the instrument panel, two in front of the pilot and two in front of the copilot. Left-hand



1. Emergency Hydraulic Shut-Off Valve
2. Utility Receptacle — Electrical
3. Copilot's Mixer Panel
4. Emergency Hydraulic Pressure Gage
5. Copilot's Collective Pitch Control
6. Copilot's Shoulder Harness Inertia Reel Lock Lever
7. Emergency Hydraulic Pump Lever
8. Landing Gear Emergency Valve Switch

Figure 1—8. Pilot's Compartment — Left Side

eters (20 and 32, figure 1—12) indicate left engine and apparent rotor rpm. Right-hand tachometers (21 and 33, figure 1—12) indicate right engine and apparent rotor rpm. The instrument face is marked off in increments of 100 from zero to 4000. This scale is calibrated for engine rpm. Range markings (figure 5—1) are superimposed on the instrument face glass. The pointers, marked 1 or 2, indicate rpm of the left or right engine respectively; pointers, marked R, indicate apparent rotor rpm. The dual tachometers are connected through independent electrical circuits to three tachometer-generators driven by the main gear box and the left and the right engine respectively. The rotor pointer indicates apparent rpm which must be divided by 14.01 to give the actual rotor rpm. For the relationship between apparent and actual rotor rpm, refer to figure A—1.

MANIFOLD PRESSURE GAGES.

Two dual manifold pressure gages (35 and 54, figure 1—12) are mounted on the instrument panel, one in front of the pilot and one in front of the copilot. The two pointers, marked 1 and 2, are connected to pressure transmitters located in each nacelle and connected to the blower case of the left and right engines respectively and indicate the intake manifold pressure in inches of mercury. The circuits operate on current from the No. 1 inverter 26-volt alternating current bus and are protected by fuses, marked MANIFOLD PRESS 1 and 2, located on the instrument fuse panel.

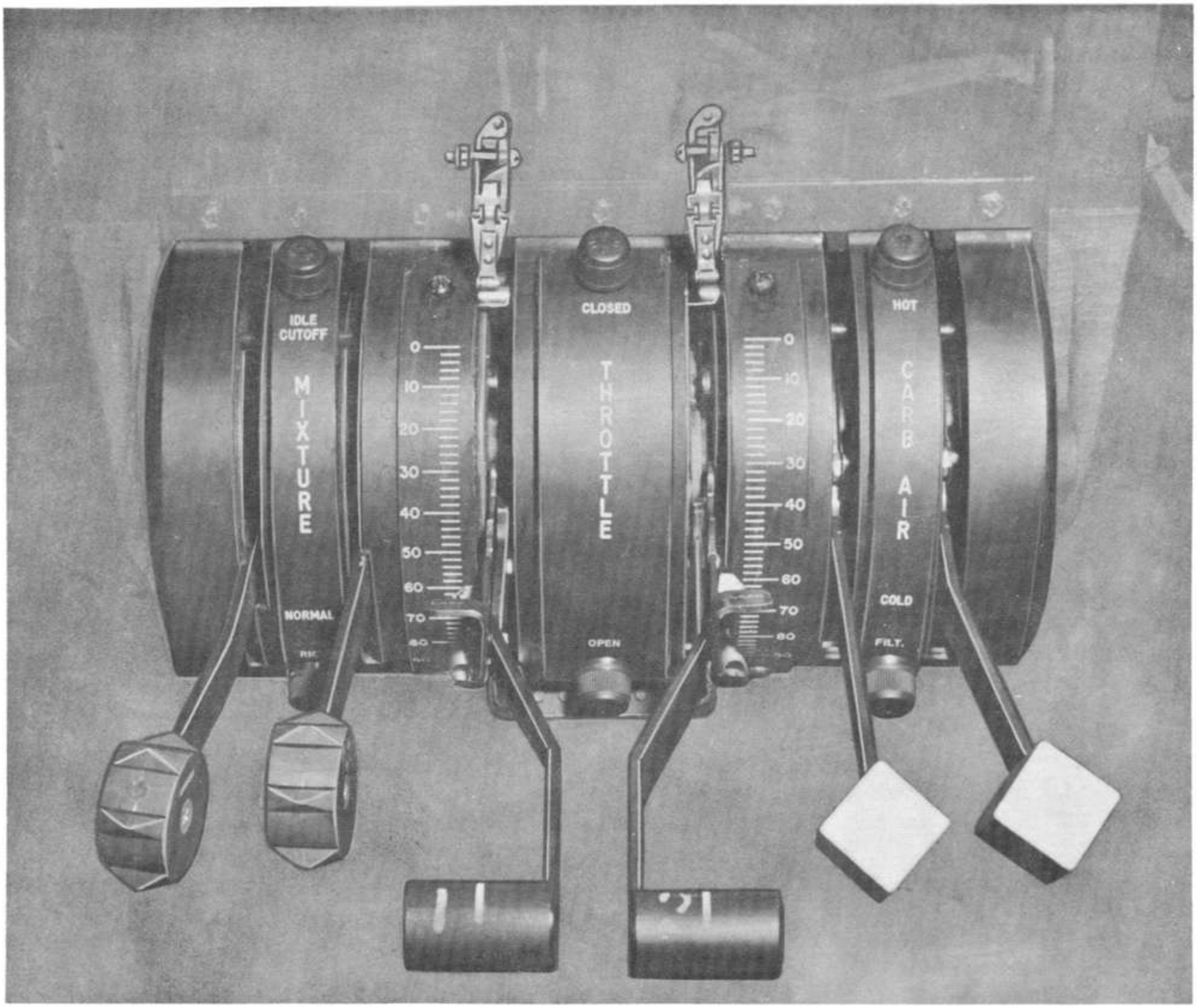


Figure 1–9. Engine Control Quadrant

CYLINDER HEAD TEMPERATURE GAGES.

Two cylinder head temperature gages (7, figure 1–12) are located on the instrument panel between the pilot and copilot to indicate right and left engine cylinder head temperatures in degrees Centigrade. The gages are connected by wiring to bayonet-type thermocouples located on the hottest operating cylinder of each engine. The circuits operate on direct current from the secondary bus and are protected by circuit breakers, marked CYL 1 and 2, under the heading ENGINE TEMPERATURES, located on the overhead circuit breaker panel.

CARBURETOR AIR TEMPERATURE GAGES.

Two carburetor air temperature gages (27, figure 1–12), one for each engine, are located on the instrument panel between the pilot and copilot. The gages are connected by resistance-element type bulbs to the induction system at the carburetors and measure the intake air temperature in degrees Centigrade. The circuits operate on direct current from the secondary bus and are protected by circuit breakers, marked CARB AIR 1 and 2, under the heading ENGINE TEMPERATURES, located on the overhead circuit breaker panel. Carburetor air temperature may be regulated by the carburetor air levers on the control quadrant.

ENGINE OIL TEMPERATURE GAGES.

Two engine oil temperature gages (9, figure 1–12), one for each engine, are located on the instrument panel. The gages are wired to a temperature bulb adjacent to the oil inlet port of each engine, and indicate engine oil temperature in degrees Centigrade. The circuits operate on direct current from the secondary bus, and are protected by circuit breakers, marked OIL INLET 1 and 2, under the heading ENGINE TEMPERATURES, located on the overhead circuit breaker panel.

ENGINE OIL PRESSURE GAGES.

Two engine oil pressure gages (8, figure 1–12) are mounted above the oil temperature gages on the instrument panel. The gages are wired to an oil pressure transmitter located in each nacelle and connected to the oil inlet pressure port on each engine. The gages are graduated in pounds per square inch. The circuits operate on current from the No. 1 inverter 26-volt alternating current bus and are protected by fuses, marked OIL PRESSURE STAGE 1 and 2, located on the instrument fuse panel.

FUEL PRESSURE GAGES.

Two fuel pressure gages (26, figure 1–12), one for each engine, are located on the instrument panel. The gages are wired to a pressure transmitter located in each nacelle and connected at the fuel inlet port of each carburetor. The gages indicate fuel pressure in pounds per square inch. The circuits operate on current from the No. 1 inverter 26-volt alternating current bus and are protected by fuses, marked FUEL PRESS 1 and 2, located on the instrument fuse panel.

TRANSMISSION SYSTEM.

The transmission system (figure 1–13) consists of three gear boxes with connecting shafting, two hydro-mechanical clutches, and auxiliary drives. The purpose of the transmission system is to reduce engine speed for driving the main and tail rotors. A hydro-mechanical rotor clutch between each engine and the main gear box permits the engines to be started, stopped, or operated while completely disengaged from the transmission system. In the event engine rpm decreases below that of the transmission system, automatic freewheeling units permit the main and tail rotors to autorotate without drag from the engines. A main drive shaft extends inboard from each clutch to drive the main gear box, which transmits engine torque upward to drive the main rotor, and aft through the tail rotor drive shaft to the intermediate gear box. From there, the pylon drive shaft extends upward to the tail gear box which drives the tail rotor. All shafts are equipped with rubber couplings to reduce shock loads. A rotor brake is located on the tail drive shaft just aft of the main gear box. An automatic shaft coupling, to permit folding of the pylon, is located aft of the intermediate gear box.

MAIN GEAR BOX. The main gear box (figure 1–13) is mounted in the center section of the wing above the cabin. The main gear box contains a two-stage planetary gear system which reduces engine rpm approximately 14 to 1 for driving the main rotor. The tail rotor is also driven through the main gear box. The tail rotor drive shaft extends aft from the lower housing of the gear

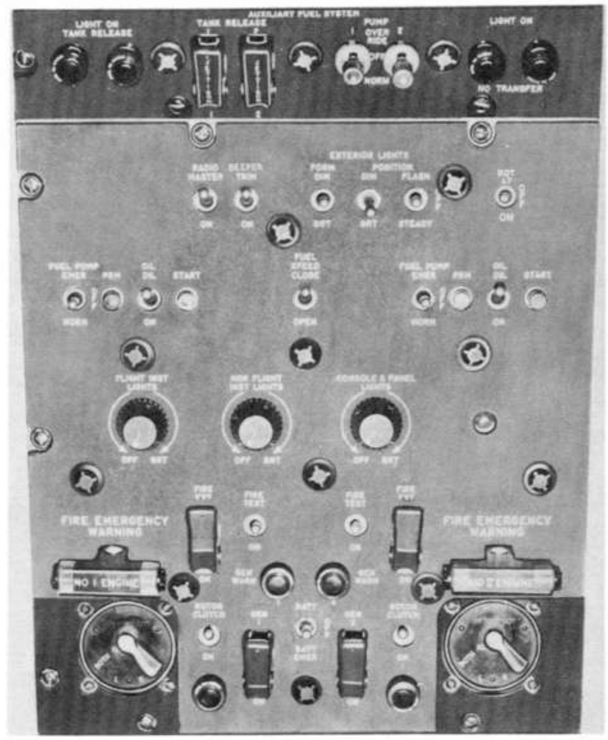


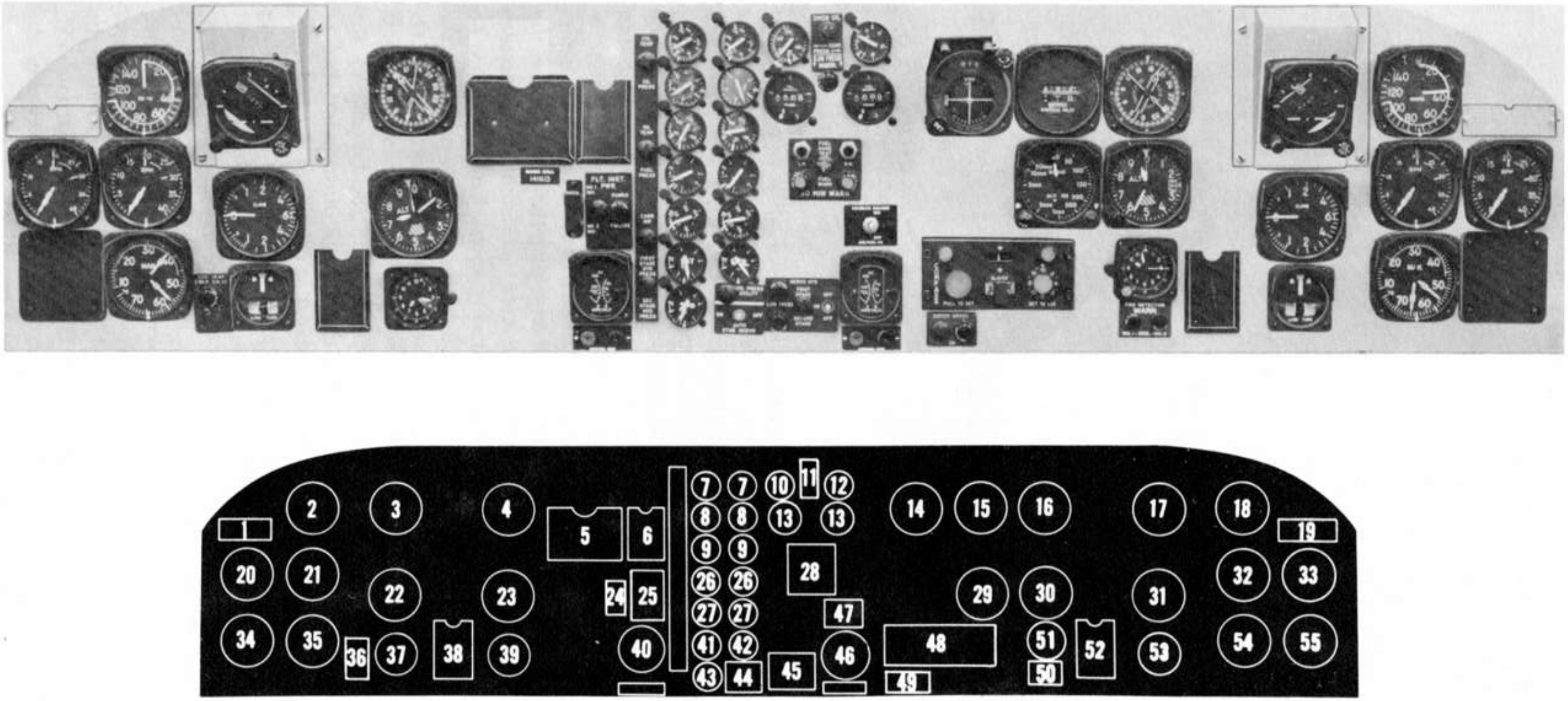
Figure 1–10. Overhead Switch Panel

box. In addition to driving the main and tail rotors, the main gear box drives the first stage servo hydraulic pump, the rotor tachometer-generator, and the main gear box oil pump, all located at the rear of the lower housing. The main gear box has its own pressure oil system for lubrication and cooling, with a capacity of approximately 6.7 gallons. A green heat stripe is located on the right side of each input housing on the main gear box. If the bearings within the housings overheat, the stripes will scorch or blister.

INTERMEDIATE GEAR BOX. The intermediate gear box (figure 1–13), located at the base of the tail rotor pylon, contains a bevel gear direct drive system to change the direction of the shafting that transmits engine power to the tail rotor. The intermediate gear box is splash-lubricated with a capacity of approximately 0.5 gallon of oil. Screened cooling air openings permit the gear box to be cooled by the main rotor down-wash. The intermediate gear box is located in the aft fuselage section with the pylon fold disconnect coupling aft of it. A green heat stripe is located on the left side of the intermediate gear box. If the bearings within the gear box overheat, the stripe will scorch or blister.

TAIL GEAR BOX. The tail gear box (figure 1–13),

Figure 1—11 deleted.



1. Airspeed Correction Chart
2. Airspeed Indicator
3. Attitude Indicator
4. Radio Magnetic Indicator
5. UHF Channelization Card Holder
6. Standby Magnetic Compass Deviation Card Holder
7. Cylinder Head Temperature Gages
8. Engine Oil Pressure Gages
9. Engine Oil Temperature Gages
10. Main Gear Box Oil Temperature Gage
11. Main Gear Box Oil Low Pressure Warning Light
12. Main Gear Box Oil Pressure Gage
13. Fuel Quantity Gages
14. Course Indicator
15. Range Indicator
16. Radio Magnetic Indicator
17. Attitude Indicator
18. Airspeed Indicator
19. Airspeed Correction Card
20. Tachometer – Left Engine and Rotor
21. Tachometer – Right Engine and Rotor
22. Vertical Velocity Indicator
23. Altimeter
24. Flight Instrument Power Switch
25. Flight Instrument Power Failure Warning Light
26. Fuel Pressure Gages
27. Carburetor Air Temperature Gages
28. Fuel Gage Test Switches and Low Level Warning Lights
29. Radar Height Indicator
30. Altimeter
31. Vertical Velocity Indicator
32. Tachometer – Left Engine and Rotor
33. Tachometer – Right Engine and Rotor
34. Turn-and-Slip Indicator (Provisions)
35. Manifold Pressure Gage
36. Landing Gear Emergency Valve Warning Light
37. Turn-and-Slip Indicator
38. Compass Correction Card Holder
39. Clock
40. Volt-Ammeter (Left Generator)
41. First Stage Servo Hydraulic Pressure Gage
42. Utility Hydraulic Pressure Gage
43. Second Stage Servo Hydraulic Pressure Gage
44. Automatic Stabilization Servo Shut-Off Switch
45. First and Second Stage Servo Shut-Off Switches and Low Pressure Warning Lights
46. Volt-Ammeter (Right Generator)
47. Squelch – Disable Switch (AN/ARC-44)
48. Gyro-Magnetic Compass Control Panel
49. Rotor Brake Warning Light
50. Fire Detector Warning Light
51. Clock
52. Compass Correction Card Holder
53. Turn-and-Slip Indicator
54. Manifold Pressure Gage
55. Turn-and-Slip Indicator (Provisions)

Figure 1–12. Instrument Panel (Typical)

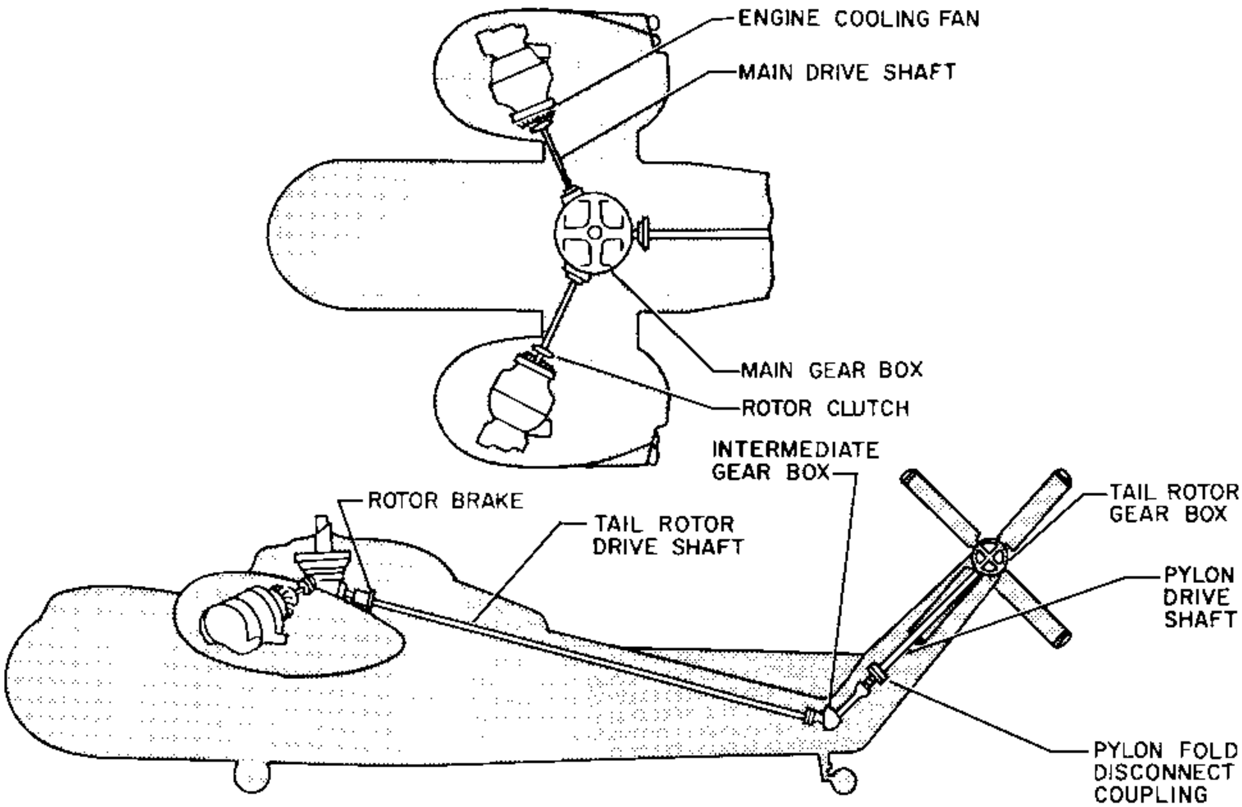


Figure 1—13. Transmission System

located at the upper end of the tail rotor pylon, contains a bevel gear reduction drive system to transmit engine torque to the tail rotor. The tail gear box also contains part of the tail rotor pitch change linkage which extends through the hollow gear box output shaft to the tail rotor head. The tail gear box is splash-lubricated with a capacity of approximately 1 gallon. A green heat stripe is located on the top side of the tail gear box. If the bearings within the gear box overheat, the stripe will scorch or blister.

CAUTION

If the heat stripe on any gear box scorches or blisters, it indicates a malfunction in that gear box. Do not resume flight until the cause is determined and corrected.

ROTOR CLUTCHES. Two hydro-mechanical rotor clutches (12, figure 1—2), one for each engine, are installed between the engines and the main gear box. One end of the clutch is bolted to the engine cooling fan hub at the engine shaft, and the other end is connected by a rubber coupling to the outboard end of the main drive shaft. Each clutch is composed mainly of a fluid coupling and a mechanical coupling. The functions of the fluid coupling are: to permit the engine to be started and operated at any speed while completely disengaged from the transmission system, and to provide a smooth and rapid acceleration of the transmission system during clutch engagement. The mechanical coupling provides a direct mechanical drive between the engine and transmission upon engagement. A freewheeling unit, which is part of the mechanical coupling, eliminates engine drag on the transmission system during autorotation by automatically disengaging the engine from the transmission system whenever main rotor speed exceeds the equivalent engine driving speed. Each clutch has its own oil systems located behind removable panels on the inboard sides of the engine nacelles. A reservoir (11, figure 1—45), with a capacity of approximately 2 US gallons, is connected by tubing to the housing of the fluid coupling. A filler cap, with a dipstick attached to check the oil level, is located at the top of the reservoir. Oil is pumped from the reservoir into the fluid coupling by a rotor clutch pump. The rotor clutch pumps operate on direct current from the primary bus and are protected by cicuit breakers, marked ENG CLUTCH 1 and 2, located on the overhead circuit breaker panel. Switches in the pilot's compartment actuate the pumps. For a more detailed explanation of the operation of the clutches, refer to ROTOR CLUTCHES, Section VII. For the specification of the oil used for servicing the clutch reservoirs, refer to figure 1—45.

ROTOR CLUTCH PUMP SWITCHES AND WARNING LIGHTS.

Two rotor clutch pump switches (figure 1–10), marked ROTOR CLUTCH, located on the forward end of the overhead switch panel, electrically control the rotor clutch pumps. A red press-to-test warning light forward of the switch lights whenever the clutch switch is in the ON position. When either switch is placed in the ON position, after the engines have been started, oil is pumped into the fluid coupling to accelerate the transmission system before mechanical engagement can be accomplished. The primary bus direct current circuit to the switch is interlocked in such a way that the rotor clutch pump switches receive no power unless the main rotor blade control lockpins are retracted after unfolding, the pylon is locked in the unfolded position, the rotor brake is off, and the tail rotor blades are unconed. It is also impossible to start folding the blades or pylon if the clutches are engaged.

ROTOR BRAKE.

A hydraulic power rotor brake, utilizing fluid from the utility hydraulic reservoir, is used to stop the rotation of the rotors and to prevent rotation of the rotors when the helicopter is parked. The brake cylinder is mounted in the pilot's compartment ceiling and the brake disc is mounted on the tail rotor drive shaft aft of the main gear box. A small accumulator (7, figure 1–45), charged with air to 300 psi, prevents a surge in hydraulic pressure when the rotor brake is applied.

ROTOR BRAKE LEVER.

The rotor brake lever (figure 1–14), which connects directly to the rotor brake hydraulic cylinder, is located on the pilot's compartment ceiling to the right of the overhead switch panel. To apply the rotor brake, the lever is pulled down and swung forward. A rotor brake warning light, located on the instrument panel, will come on when the rotor brake is applied. The rotor brake lever is automatically locked in the applied (forward) position by a spring-loaded rotor brake lever lockpin (3, figure 1–7), located at the forward inboard side of the cylinder. The lockpin is equipped with a small toggle handle which points inboard in line with the pin when the rotor brake lever is locked. This handle must be pushed forward or aft to extract the lockpin before the rotor brake lever may be released. The lockpin is free to rotate in its housing; if the toggle handle will not actuate in a fore-and-aft direction, push it up or down. The lockpin handle may be left in the aft position to render the lockpin inoperative, or it may be returned to the neutral in-line position to allow the lockpin to retract into the housing where it will automatically lock the rotor brake lever at the next application. When the rotor brake is applied, hydraulic pressure in the line actuates a pressure switch which breaks the circuit to the rotor clutch pump switches, making it impossible to engage the clutches. To release the rotor brake, the lever is unlocked, then pulled aft and swung upward where it is locked in a detent. If the lever is not held securely in the OFF (up) position by the detent, the weight of the lever will cause the rotor brake to drag. A dragging rotor brake, besides cutting down effective engine power, will soon wear itself out. The rotor brake is designed to bring the rotors to a dead stop from hovering rpm within 15 seconds, although normally the brake should be applied more gradually. Do not apply the brake while the rotors are being engine-driven.

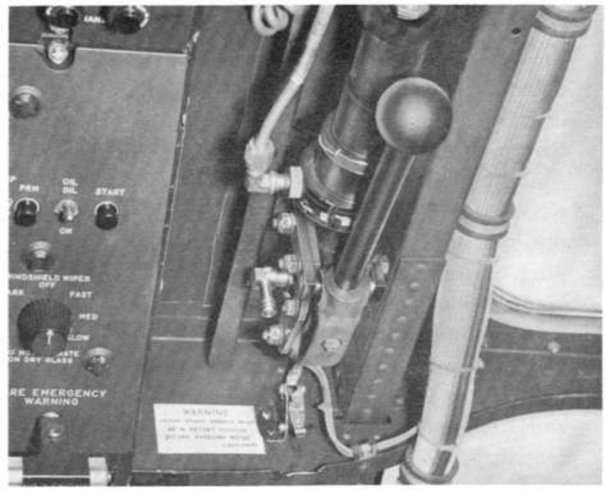


Figure 1–14. Rotor Brake Lever and Lockpin

ROTOR BRAKE WARNING LIGHT.

A red press-to-test rotor brake warning light (49, figure 1–12) is located on the instrument panel below the gyro-magnetic compass control panel. When the rotor brake lever is moved down and forward to the ON position, hydraulic pressure in the line actuates a pressure switch which allows direct current from the secondary bus to light the rotor brake warning light.

ROTOR SYSTEM.

The rotor system consists of the main rotor system and the tail rotor system. Both systems are driven by the engines and transmission system and controlled by the flight control system. The main rotor blades and the pylon may be folded to conserve space; refer to **MAIN ROTOR BLADE AND PYLON FOLDING,** Section IV.

MAIN ROTOR SYSTEM. The main rotor system consists of the main rotor blades, the main rotor assembly (10, figure 1–2), and the linkage necessary to transmit main rotor flight control movement to the blades. The main rotor assembly, consisting of a main rotor hub assembly and a star assembly, is mounted directly above the main gear box and is splined to the main gear box drive shaft. The five all-metal main rotor blades are hinged at the main rotor hub assembly in such a way that each blade is free to flap vertically, hunt horizontally, and rotate about its span-wise axis to change the angle of incidence. Droop restrainers, attached to the hub, limit the downward movement, and anti-flapping restrainers limit the upward movement of the blades about the flapping hinges when the blades are stopped or turning at low speed. When rotor speed is increased to approximately 700 apparent rpm, centrifugal force automatically releases the droop restrainers, and the blades are supported by the combination of centrifugal force and lift. At 980 to 1120 apparent rpm, centrifugal force automatically releases the anti-flapping restrainers. Hydraulic dampers, connected between each blade and the rotor hub, minimize hunting movement of the blades about the vertical hinge as they rotate, prevent shock to the blades when the rotor is started or stopped, and aid in the prevention of ground resonance. The angle of incidence, or pitch, of the main rotor blades is controlled by the main rotor flight control system which connects to the blades through a star assembly at the lower section of the main rotor head. The star assembly consists of an upper star rotated by the main rotor drive shaft and a lower star which is stationary. Both stars are mounted on a ball-ring and socket assembly which keeps them parallel, but allows them to be tilted, raised, or lowered simultaneously by the main rotor flight control system which connects to arms on the lower (stationary) star assembly. Linkage on the rotating star transmits the control motion to the blades. The five all-metal main rotor blades are constructed of aluminum alloy with the exception of forged steel cuffs which attach the root ends of the blades to the sleeve-spindle assemblies on the main rotor hub assembly. Two types of rotor blades are utilized in the field, a wide chord blade and a narrow chord blade. The wide chord blade (23.65-inch chord) consists of a hollow extruded aluminum alloy spar with an 8-degree negative twist. Individual pockets of ribbed construction are bonded to the spar to form the trailing edge. The narrow chord blade (21.5-inch chord) consists of a hollow extruded aluminum alloy spar with 14 degrees of negative twist. The trailing edge consists of individual pockets of honeycomb and ribbed core construction bonded to the leading edge spar.

TAIL ROTOR SYSTEM. The tail rotor system consists of four all-metal rotor blades, the tail rotor assembly (1, figure 1–2), and the pitch-change mechanism. The tail rotor hub is splined to the tail gear box drive shaft which transmits engine torque to the blades. The blades are attached to the tail rotor hub in such a way that they are free to flap, hunt, and rotate about their axis for pitch variation. They are also connected to each other by shock absorbers which help to distribute varying air loads and reduce vibration. The blade pitch changing mechanism transmits tail rotor flight control movement through the hollow tail rotor drive shaft to the blades. Each tail rotor blade consists of a single ribbed trailing edge pocket bonded to a hollow extruded leading edge spar.

TAIL ROTOR BLADES CONING SYSTEM.

To prevent the tail rotor blades from flapping when the helicopter is parked on the ground, the tail rotor blades are coned against their stops by an electrically operated coning ring located on the tail rotor gear box. The tail rotor blades must be unconed before positioning the rotor blades for main rotor blade folding. The blades are then coned prior to folding of the pylon. Power from the secondary bus is available for blade coning only if the main rotor is turning less than 6 rpm (transmission low pressure warning light on) and neither rotor clutch pump is turned on. An electrical interlock will also prevent engaging either rotor clutch if the tail rotor blades are coned, as shown by the lighting of a warning light. The tail rotor blades coning system operates on direct current from the secondary bus and is protected by a circuit breaker, marked TAIL ROTOR CONING, located on the overhead circuit breaker panel.

TAIL ROTOR CONING SWITCH AND WARNING LIGHT.

A tail rotor coning switch (figure 4–27), marked TAIL ROTOR CONING, with two marked positions, CONED and FLT POSIT, is located on the left side of the control console. When the switch is placed in the CONED position, the blades will cone and the warning light, marked BLADES CONED, will be turned on. A limit switch will turn off the actuator when coning is accomplished but the warning light will remain on. When the switch is placed in the FLT POSIT position, the tail rotor blades uncone; a limit switch will turn off the actuator and the warning light when unconing is accomplished.

Note

When the tail rotor coning switch is placed in the FLT POSIT position, visually check the tail rotor coning ring.

ENGINE OIL SYSTEM.

The oil system for each engine includes a bladder-type oil tank, located on the outboard side of each nacelle, and an oil cooling system below the tank. The oil tank

(10, figure 1—45), with a normal quantity of approximately 13.3 gallons, is filled through a hinged door on the top outboard side of the nacelle. For extended flights, the tank may be filled to 17 gallons. A dip stick to check the oil level is located forward of the filler neck. Engine oil flows from the bottom of the tank to a hand-operated drain valve (figure 1—15), located on the outboard side of each wheel well. The oil system may be drained by opening this valve. From the drain valve, tubing leads forward to the fire wall where oil flow to the engine may be cut off in the event of fire by a valve actuated by the engine fire emergency handle on the overhead switch panel. Flexible hose leads from the fire wall to the engine oil inlet port. After circulating through the engine, the oil is fed aft through the fire wall to the oil cooling system. An oil dilution system enables the oil to be diluted for cold weather starting.

ENGINE OIL COOLING SYSTEM.

The engine oil cooler for each engine is mounted horizontally below each oil tank. Cooling air enters a duct at the bottom of the nacelle and is led upward through the horizontally mounted oil cooler. Above the oil cooler the air enters a chamber and is then let out the aft end of the nacelle through two engine exhaust ejector ducts (24, figure 1—2). Engine exhaust enters the forward end of the chamber and is expelled through the same ejector ducts. The exhaust creates a partial vacuum and accelerates the flow of air through the oil cooler. The oil cooling system provides a positive flow of cooling air whenever the engine is running. A thermostatic valve at the oil cooler inlet port controls the circulation of oil returning from the engine. For the specification and grade of engine oil, refer to figure 1—45. Oil dilution procedure is shown in section IX.

OIL DILUTION SYSTEM.

The oil dilution system supplies fuel from the fuel pressure indicator lines to the engine oil inlet lines for the purpose of decreasing oil viscosity at low ambient temperatures. Prior to oil dilution, the oil tank should not contain more than 13.3 gallons of oil. For each engine, the flow of fuel is controlled by a solenoid valve which is operated by an oil dilution switch. The solenoid valve operates on direct current from the secondary bus, and is protected by a circuit breaker, marked ENG START, PRM, and OIL DIL, located on the overhead circuit breaker panel. A manually operated shut-off valve is installed in the oil dilution line above the oil system drain valve. For oil dilution procedure and oil dilution time, refer to section IX.

OIL DILUTION SWITCHES.

Two spring-loaded oil dilution switches (figure 1—10), marked OIL DILUTION, one for each engine, are located on the overhead switch panel. When the switch is held in the ON position, the electrically operated solenoid admits fuel to the oil inlet line for oil dilution. When the switch is released, it automatically returns to the OFF (center) position and oil dilution stops.

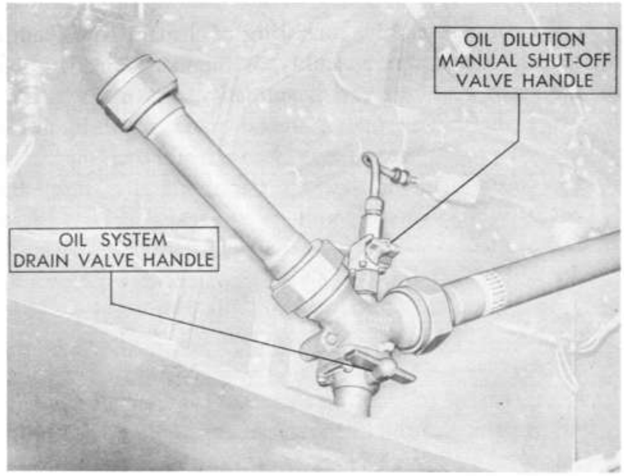


Figure 1—15. Oil Dilution Manual Shut-Off Valve Handle

OIL DILUTION MANUAL SHUT-OFF VALVE HANDLE.

An oil dilution manual shut-off valve handle (figure 1—15) is located in each oil dilution line on the outboard side of each wheel well, above the Y oil drain. This handle should be kept closed (horizontal position) to prevent possible dilution of engine oil due to a leaking oil dilution solenoid valve. The handle must be opened (vertical position) before oil dilution can be accomplished by actuating the oil dilution switch. Should the manual shut-off valve be left in the closed position, it will be indicated by a steady fuel pressure reading instead of a decrease when commencing oil dilution.

TRANSMISSION OIL SYSTEMS.

Each of the three transmission gear boxes has an individual oil system. The main gear box is pressure-lubricated and the intermediate and tail gear boxes are splash-lubricated. For the specification and grade of the oil used in the various gear boxes, refer to figure 1—45.

MAIN GEAR BOX OIL SYSTEM. Pressure for lubricating the main gear box is supplied by a pump mounted on the gear box lower housing. Oil is pumped from the gear box sump through a hose to an oil cooler (3, figure 1—3), located behind the main gear box. Cooling air enters the forward end of the main gear box fairing through a screened main gear box oil cooling air intake (11, figure 1—2) and is forced through the oil cooler by a blower driven by belts from the tail drive shaft. The air is then exhausted through a screened main gear box oil cooling air outlet (9, figure 1—2) at the rear of

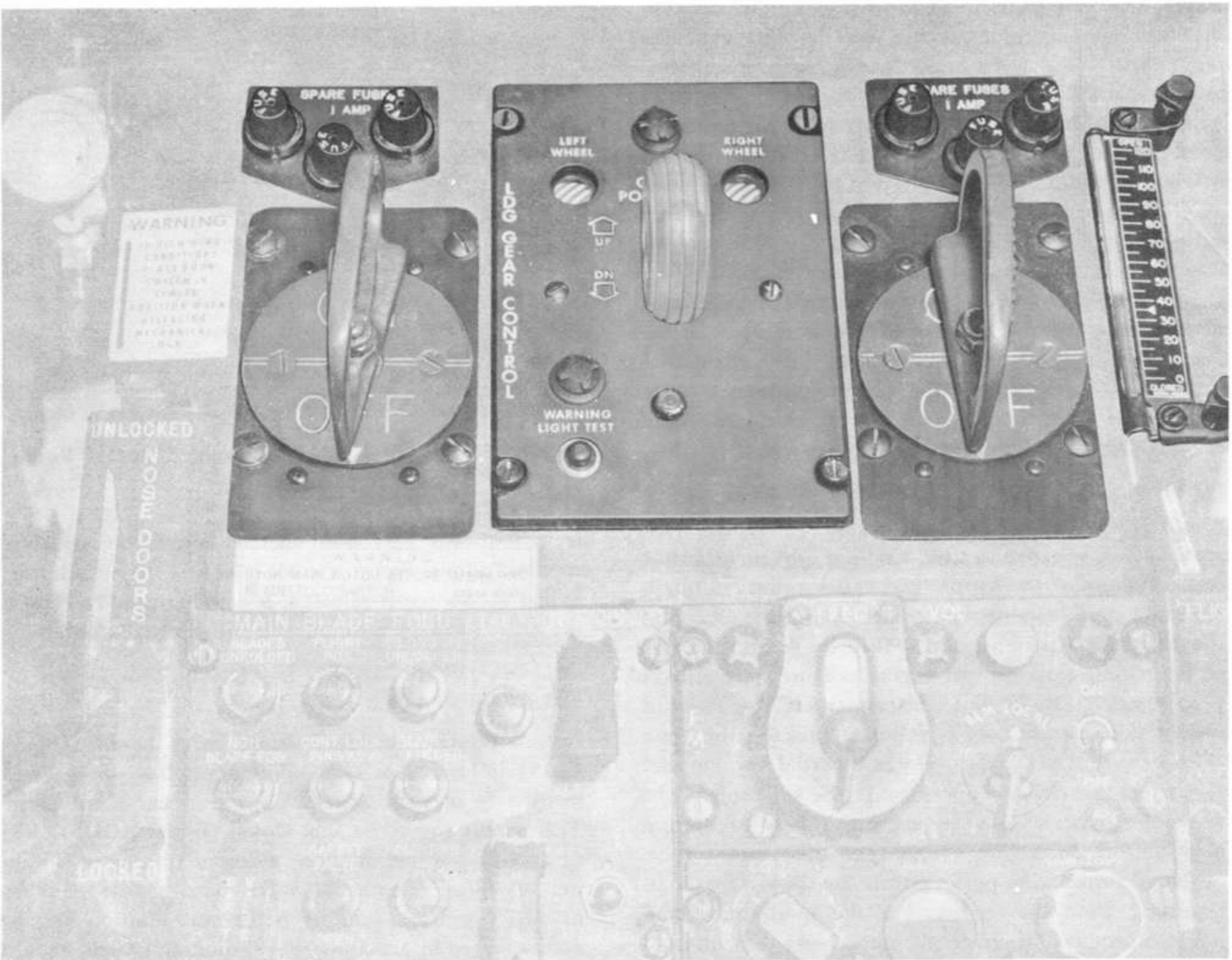


Figure 1–16. Control Console — Forward Section

the fairing. After passing through the oil cooler, the oil returns to the main gear box where it is sprayed onto the gears and bearings through jets built into the gear box castings. An oil filler, accessible from the left side of the main rotor fairing, is located on the left side of the gear box. A window in the gear box below the oil filler, visible from the cabin on the left aft side, provides a sight check for the oil level in the main gear box. An inspection light is located next to the sight gage. The light is controlled by a parking and gear box inspection light switch (figure 4–10) inside the cabin, forward of the cargo door. Oil capacity is approximately 6.7 gallons.

MAIN GEAR BOX OIL PRESSURE GAGE AND WARNING LIGHT.

The main gear box oil pressure gage (12, figure 1–12) is located on the instrument panel. The gage is graduated in pounds per square inch, and is actuated by a pressure transmitter connected to the gear box oil inlet port. The circuit operates on current from the No. 1 inverter 26-volt alternating current bus and is protected by a fuse, marked XMSN, located on the instrument fuse panel. The main gear box oil low pressure warning light (11, figure 1–12), marked XMSN OIL LOW PRESS. WARN, is located at the left of the pressure gage. The light, of the press-to-test type, operates on direct current from the secondary bus. The warning light will come on when below normal pressure is sensed at the last pressure jet in the main gear box lubrication system.

WARNING

If the main gear box oil pressure gage reads less than 25 psi, or the main gear box oil low pressure warning light goes on, it indicates a loss of pressure in the system and improper lubrication of the main gear box. An immediate landing should be made and flight should not be resumed until the trouble is corrected.

MAIN GEAR BOX OIL TEMPERATURE GAGE.

The main gear box oil temperature gage (10, figure 1–12), located on the instrument panel, is graduated in degrees Centigrade. The gage is wired to an oil temperature bulb located adjacent to the main gear box oil outlet port. The circuit operates on direct current from the secondary bus and is protected by a circuit breaker, marked XMSN OIL TEMP, located on the overhead circuit breaker panel.

WARNING

If the main gear box oil temperature gage reads more than 140°C (280°F), it indicates a malfunction in the main gear box or in the oil cooling system, and lack of proper lubrication in the main gear box. An immediate landing should be made and the flight should not be resumed until the trouble is corrected.

INTERMEDIATE AND TAIL GEAR BOX OIL SYSTEMS.

Both the intermediate and the tail gear boxes are splash-lubricated from individual sump systems. Internal spiral channels insure oil lubrication to all bearings. An oil filler plug, drain plug, and oil level window are located in each gear box casting. A light, operated from the battery bus, is located beside each window to aid in oil level inspection. All gear box oil level inspection lights are controlled by a parking and gear box inspection light switch (figure 4–10), located inside the cabin, forward of the cargo door. Oil capacity for the intermediate gear box is about 0.5 gallon and for the tail gear box, 1 gallon.

FUEL SYSTEM.

The fuel system (figure 1–18) consists of two independent fuel systems: a right engine fuel system and a left engine fuel system. The fuel tank for each engine is located in the wing and nacelle behind the engine fire wall. Sump-mounted booster pumps in each tank supply fuel under pressure to the system. Fuel flows from the tank through a fuel strainer and shut-off valve to the engine-driven fuel pump and then to the carburetor. A cross-feed system connects the two independent systems. Fuel for the cabin heater and auxiliary power unit is drawn from the left tank. Jettisonable auxiliary fuel tanks may be installed under each wing; refer to **AUXILIARY FUEL SYSTEM** in this section. Fuel quantities for all tanks are shown in figure 1–19. Fuel specification and grade are shown in figure 1–45.

FUEL TANKS.

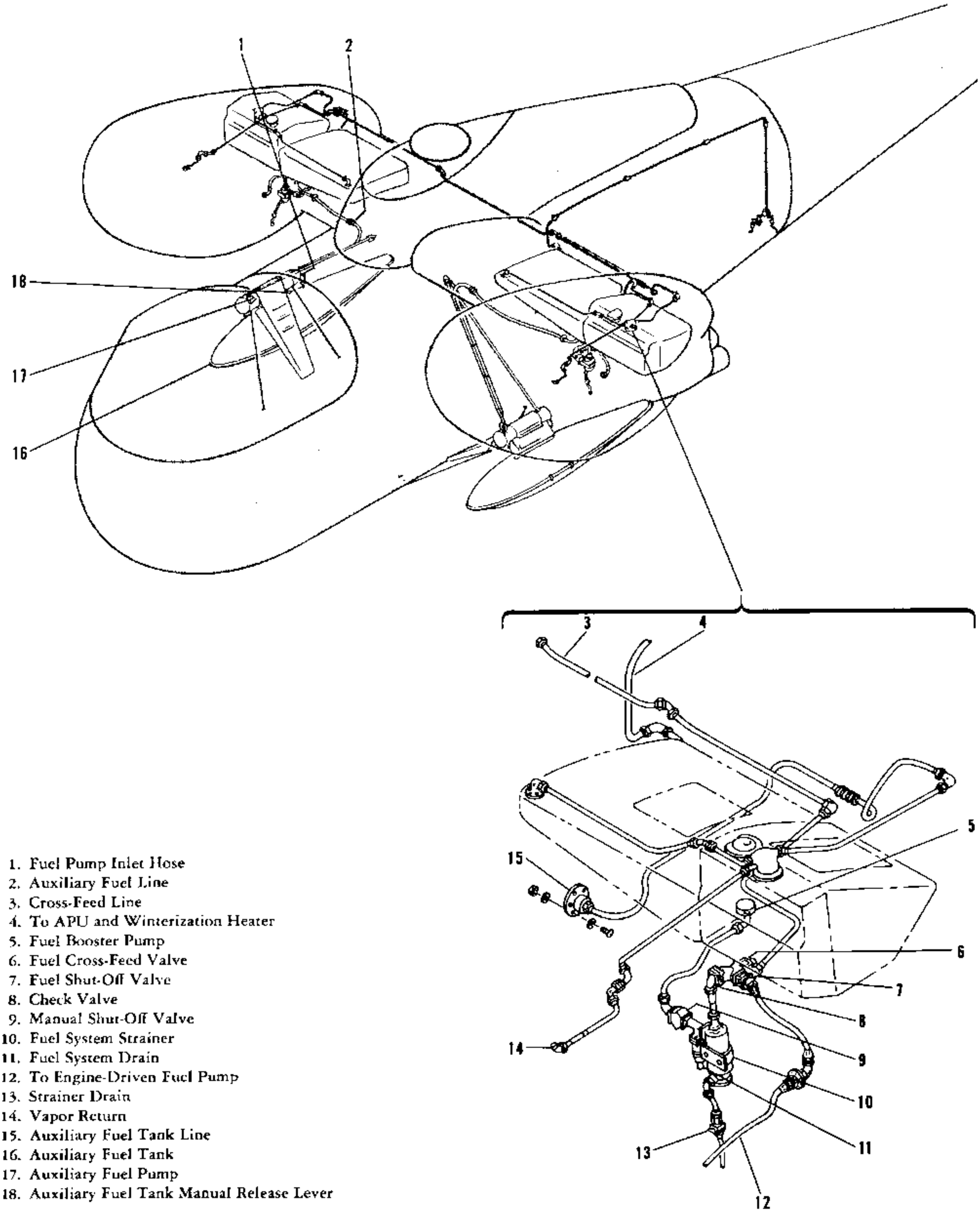
Each of the two main fuel tanks (2, figure 1–45) consists of two interconnected fuel cells, one located in the wing section and the other located in the nacelle. (Prior to BuNo 141610, the nacelle and wing cells were self-sealing; however, on BuNos 141610 and subsequent, the nacelle self-sealing tanks have been replaced with bladder-type tanks.) Both cells are filled through a fuel filler cap located on the top of the nacelle. Fuel flows by gravitational force from the nacelle cell into the wing cell. Flapper valves, in the cell interconnects, prevent reversal of flow from the wing cell to the nacelle cell during extreme helicopter attitudes. The fuel booster pumps are located in the sumps of the wing cells. A level control valve and float switch in the nacelle cells prevent fuel overflow when transferring fuel from the auxiliary tanks.

FUEL SHUT-OFF SWITCHES.

Two rotary fuel shut-off switches (figure 1–16), one for each tank, are located on the control console just aft of the instrument panel. The switches have two marked positions, ON and OFF. Each switch actuates an electrically operated fuel shut-off valve in the fuel line aft of the fire wall. The valves operate on direct current from the primary bus and are protected by a circuit breaker, marked FUEL SHUT-OFF, located on the overhead circuit breaker panel. When the switch is placed in the ON position, fuel flows from each tank to the engine in front of it; when the switch is rotated to the OFF position, fuel is cut off from the engines. The fuel shut-off valves are also actuated by the fire emergency handles located on the overhead switch panel.

FUEL CROSS-FEED SWITCH.

A two-position fuel cross-feed switch (figure 1–10), marked FUEL-X-FEED, is located on the overhead switch panel. The switch has two marked positions, CLOSE and OPEN. The switch actuates fuel cross-feed valves in the fuel lines by direct current from the primary bus and is protected by a circuit breaker, marked FUEL-X-FEED, located on the overhead circuit breaker panel. Normally, the switch is in the CLOSE position and each engine obtains fuel from the tank directly behind it. Placing the switch in the OPEN position opens valves in the cross-feed line which connects the two fuel systems. The cross-feed system may be used to supply fuel under pressure from both tanks to both engines, from either tank to both engines, or to either engine from both or either fuel tank during single-engine operation. The cross-feed system does not transfer fuel between tanks. For additional information on



1. Fuel Pump Inlet Hose
2. Auxiliary Fuel Line
3. Cross-Feed Line
4. To APU and Winterization Heater
5. Fuel Booster Pump
6. Fuel Cross-Feed Valve
7. Fuel Shut-Off Valve
8. Check Valve
9. Manual Shut-Off Valve
10. Fuel System Strainer
11. Fuel System Drain
12. To Engine-Driven Fuel Pump
13. Strainer Drain
14. Vapor Return
15. Auxiliary Fuel Tank Line
16. Auxiliary Fuel Tank
17. Auxiliary Fuel Pump
18. Auxiliary Fuel Tank Manual Release Lever

Figure No. 1—17. Fuel System (Sheet 1)

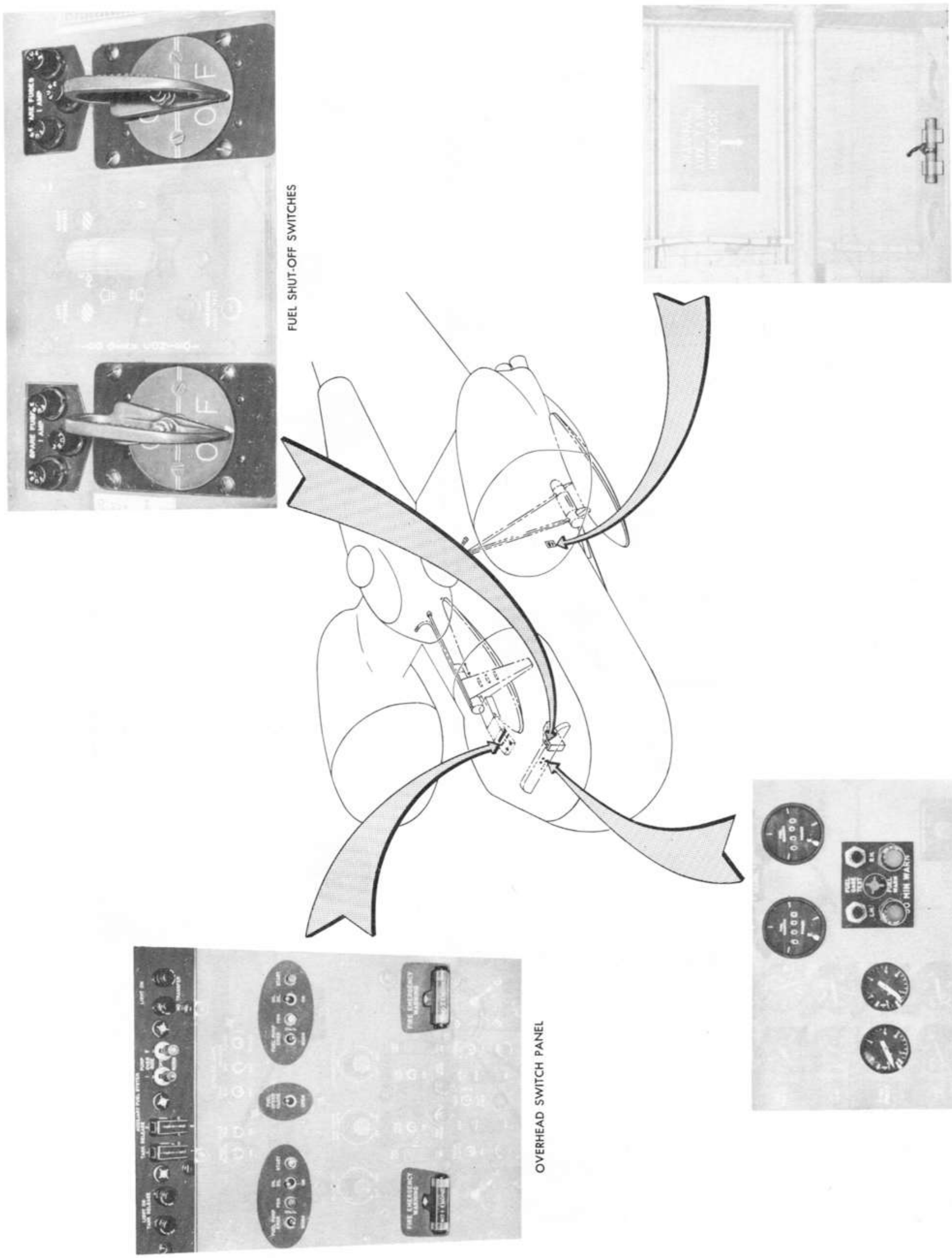


Figure No. 1—17. Fuel System (Sheet 2)

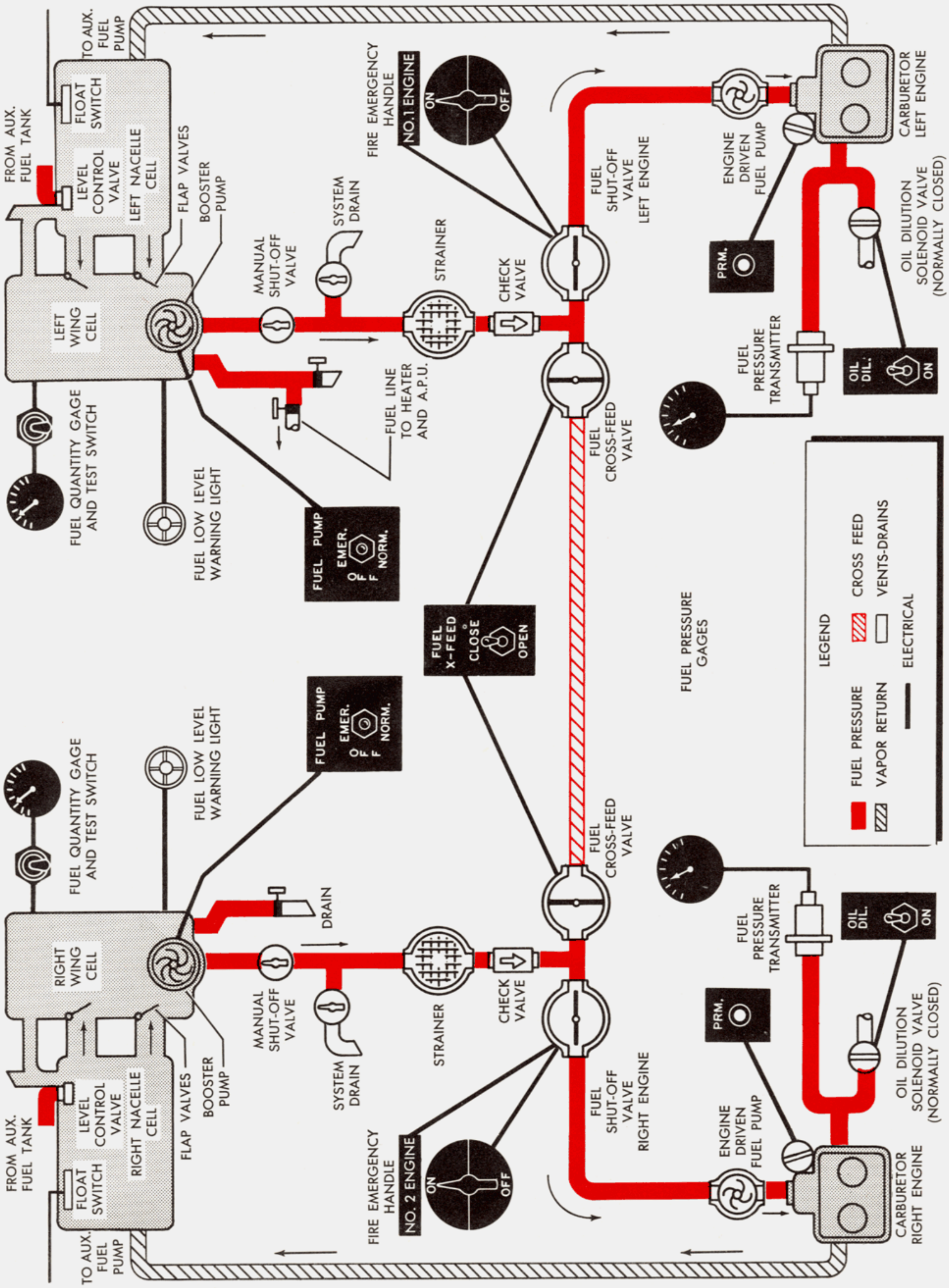


Figure 1—18. Fuel System Schematic Diagram

FUEL QUANTITY DATA								
	LEFT MAIN		RIGHT MAIN		AUX TANKS MK 8 MOD 1		AUX TANKS MK 12 MOD 1	
	GALLONS	POUNDS	GALLONS	POUNDS	GALLONS	POUNDS	GALLONS	POUNDS
FULLY SERVICED	203	1218	204	1224	300	1800	150	900
UNUSABLE	5	30	6	36	8	48	6	36
USABLE	198	1188	198	1188	292	1752	144	864

1. Total usable fuel, main tanks only — 396 gallons, 2376 pounds.
2. Total usable fuel, main tanks and 150 gal. Aux — 540 gallons, 3240 pounds.
3. Total usable fuel, main tank and 300 gal. Aux — 688 gallons, 4128 pounds.
4. Usable fuel determined at 8 degrees nose-down attitude.
5. Data compiled from actual test July 10, 1956 and January 24, 1957.

Figure 1—19. Fuel Quantity Data

this system, refer to **FUEL SYSTEM MANAGEMENT,** Section VII.

FUEL BOOSTER PUMP SWITCHES.

Two fuel booster pump switches (figure 1—10), marked FUEL PUMP, are located on the overhead switch panel. Each switch has three positions: NORM., OFF, and EMER. The NORM. position is used for starting, warm-up, and during flight, while the EMER. position is used for take-off, hovering, landing, and in the event of failure of the engine-driven fuel pumps. Should an engine-driven fuel pump fail, the booster pump in the EMER. position will deliver sufficient fuel under pressure to the carburetor. The booster pumps operate on direct current from the primary bus, and are protected by circuit breakers, marked FUEL PUMP 1 and 2, located on the overhead circuit breaker panel.

FUEL QUANTITY GAGES.

Two fuel quantity gages (13, figure 1—12), one for each tank, are located on the instrument panel. The gages are connected to two tank capacitance units in each tank through bridge circuits and an amplifier. The capacitance system of fuel quantity gaging is practically unresponsive to volumetric changes resulting from variations in temperature. No moving parts are used in the fuel tanks; instead, the dielectric properties of the fuel between two electrodes in each tank unit are utilized to furnish a measurement of fuel quantity. The voltage drop between the two electrodes will vary as the fuel level varies. The fuel gages are calibrated to measure this voltage drop in pounds of fuel. The fuel quantity indicating circuits operate on current from the No. 1 inverter 115-volt alternating current bus and are protected by fuses, marked FUEL QTY 1 and 2, located on the instrument fuse panel.

WARNING

If the power to the gages fails, they will continue to indicate the quantity of fuel shown at the time of power failure.

FUEL QUANTITY GAGE TEST SWITCHES.

The fuel quantity gages may be tested for proper functioning by means of two push-button fuel quantity gage test switches (28, figure 1—12), marked FUEL GAGE TEST, LH, RH, which are located below the fuel quantity gages. Pressing a test switch for approximately 10 seconds will induce a current reversal in the system and cause the pointer to turn counterclockwise toward zero. Upon release of the push button, the pointer should return to the previous reading. This test determines whether or not the fuel quantity indicating system is operating correctly. The test circuits operate on direct current from the primary bus and are protected by circuit beakers, marked FUEL QUANTITY 1 and 2, located on the overhead circuit breaker panel.

FUEL LOW LEVEL WARNING LIGHTS.

Two red low level warning lights (28, figure 1—12), marked FUEL WARN 30 MIN WARN, are located on the instrument panel below the fuel quantity indicators. One light for each tank will light on direct current from the primary bus when there are approximately 370 to 400 pounds of fuel remaining in the tank. This is sufficient for approximately 30 minutes of flight at maximum continuous power.

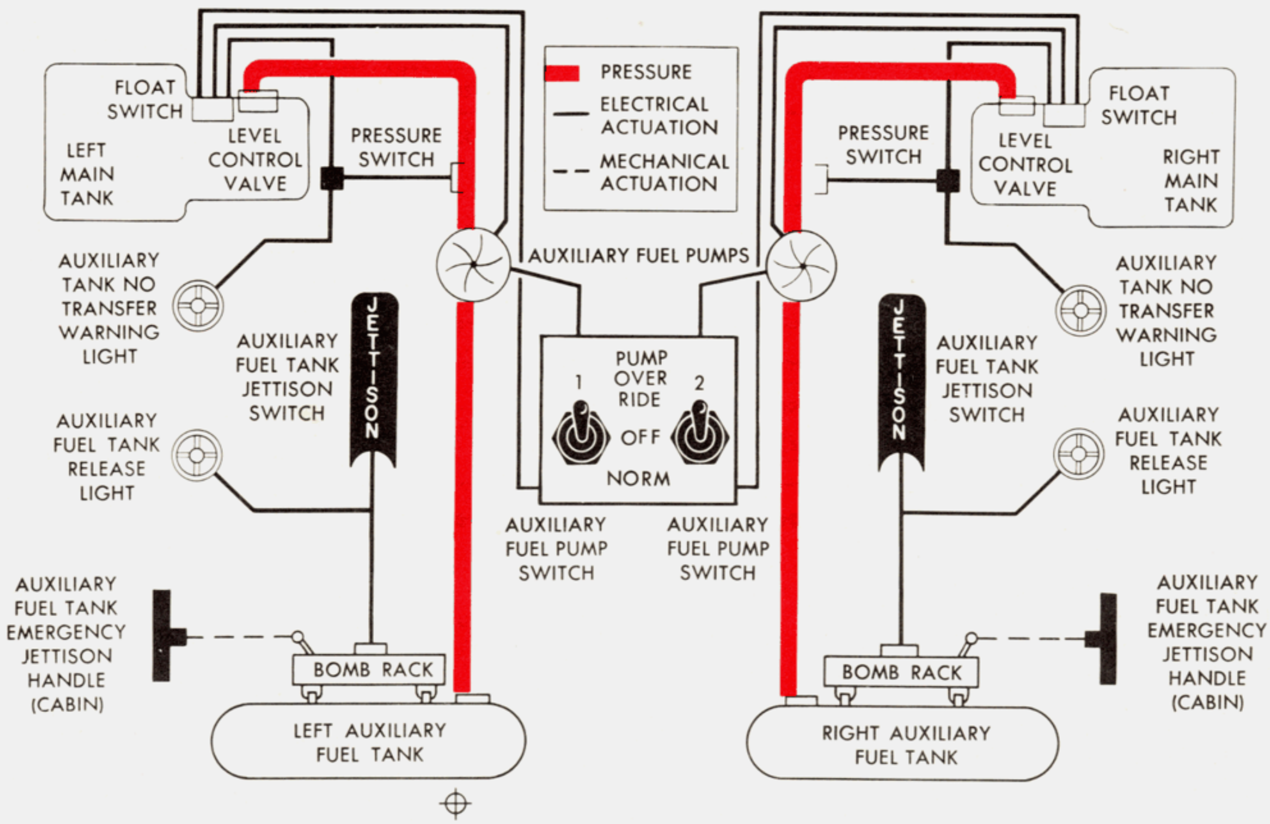


Figure 1—20. Auxiliary Fuel System Schematic Diagram

Note

During flight in rough air or rough handling of the helicopter in flight, the low level warning lights will fail to light until a level lower than 370 to 400 pounds is reached.

AUXILIARY FUEL SYSTEM.

The auxiliary fuel system (figure 1—20) consists of two detachable housings, each containing a fuel pump and a bomb rack, which are mounted, one on each side of the fuselage beneath the wing inboard of the nacelles. The housings are attached to the helicopter with struts and cables. Two types of jettisonable fuel tanks may be installed, a small tank with a capacity of 150 US gallons, and a larger tank with a capacity of 300 US gallons. Removable fuel hoses lead from the auxiliary fuel pumps to disconnect fittings on the inboard sides of the nacelles. Tubing then conducts auxiliary fuel into the top of the nacelle fuel cells of the main fuel system. Auxiliary fuel is supplied to the engines through the main fuel system. A level control valve in each nacelle cell shuts off auxiliary fuel and prevents overflowing of the main tanks, even though the auxiliary fuel pump is running; thus, it is possible to keep the main tanks full until all fuel from the auxiliary tanks is consumed. A float switch provides additional protection against overflow by shutting off the pump if the level control valve malfunctions. All auxiliary fuel system controls are located in the pilot's compartment on a panel, marked AUXILIARY FUEL SYSTEM, located adjacent to the overhead switch panel. The auxiliary fuel pumps operate on direct current from the primary bus and are protected by circuit breakers, marked AUX FUEL 1 and 2, located on the overhead circuit breaker panel.

AUXILIARY FUEL PUMP SWITCHES.

Two three-position auxiliary fuel pump switches, marked PUMPS 1, 2, located on the auxiliary fuel system panel (figure 1—10), control the operation of the auxiliary fuel pumps for each auxiliary tank. The switch, marked 1, operates the pump for the left auxiliary tank, and the switch, marked 2, operates the pump for the right auxiliary tank. When the switches are placed in the NORM. position, fuel is pumped from the auxiliary fuel tanks into the respective main fuel tank. If the float switch shuts off the auxiliary fuel pump, the NO TRANSFER warning light will come on. Fuel transfer may be continued by placing the auxiliary fuel pump switch in the OVERRIDE position which will restore electrical power to the pump. The OVERRIDE position will also allow the auxiliary fuel pump to be operated for a ground check of the fuel system when the main tanks are full enough to open the float switch. Fuel cannot be transferred from an auxiliary fuel tank to the main fuel tank on the opposite side of the helicopter. Either auxiliary fuel pump may be operated separately, if desired.

WARNING

If the OVERRIDE position is used to transfer fuel from an auxiliary fuel tank to the respective main tank, because of a malfunction of both the level control valve and the float switch, monitor the fuel quantity gage to preclude overflowing of the main tank. Fuel transfer can be controlled by periodic use of the OVERRIDE and OFF positions of the auxiliary fuel pump switch.

AUXILIARY FUEL TANK NO TRANSFER WARNING LIGHTS.

Either of two red auxiliary fuel tank no transfer warning lights (figure 1—10), located on the auxiliary fuel system panel, marked LIGHT ON NO TRANSFER, 1 (left tank), and 2 (right tank), will illuminate whenever a pressure switch, located in each auxiliary fuel tank outlet line, senses a drop in pressure below the normal operating level. A loss of pressure is normally the result of exhausting the fuel supply in the auxiliary tank; however, the light will also indicate failure of the auxiliary tank fuel pump, or the shutting off of the pump by the float switch in the nacelle cell when the tanks are overfilled. The light goes out when the auxiliary fuel pump switch is turned off.

AUXILIARY FUEL TANK JETTISON SWITCHES.

Two auxiliary fuel tank jettison switches (figure 1—10), marked TANK RELEASE, with switch guards, marked JETTISON, are located on the auxiliary fuel system panel. When a jettison switch is actuated, the tank is released. The switch, marked 1, jettisons the left tank and the switch, marked 2, jettisons the right tank.

AUXILIARY FUEL TANK MANUAL JETTISON HANDLES.

In case of electrical failure of the auxiliary tank jettison system, the auxiliary fuel tanks may be released from inside the cabin by pulling two auxiliary fuel tank jettison handles (figure 1—21), marked MANUAL AUX. TANK RELEASE, located on each side of the fuselage panel directly inboard of the auxiliary fuel system housings. The handles are stowed in spring-clips and are attached to cables that trip the manual release levers on the bomb racks and jettison the auxiliary fuel tanks.

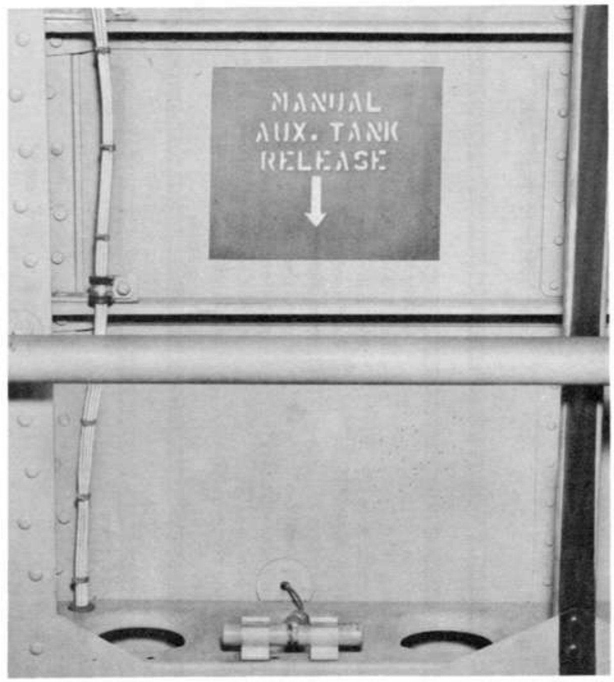


Figure 1—21. Auxiliary Fuel Tank Jettison Handle

AUXILIARY FUEL TANK RELEASED INDICATOR LIGHTS.

Either of two red auxiliary fuel tank released indicator lights (figure 1—10), marked LIGHT ON TANK RELEASED 1 (left tank), and 2 (right tank), will light when the corresponding jettisonable tank is jettisoned. The light is shut off after the tank is released when finger pressure is removed from the jettison switch.

ELECTRICAL POWER SUPPLY SYSTEM.

Electrical power is supplied by two basic systems: a 28-volt direct current system and a 115-volt alternating current system. Power sources, distribution of power, and equipment operating from each system are shown in figure 1—22.

DIRECT CURRENT POWER SUPPLY SYSTEM.

The primary sources of power for the operation of all electrical equipment are two generators. Secondary power sources are a battery, an external power receptacle, and an auxiliary power unit. Power is distributed through three dc busses: the primary, secondary, and

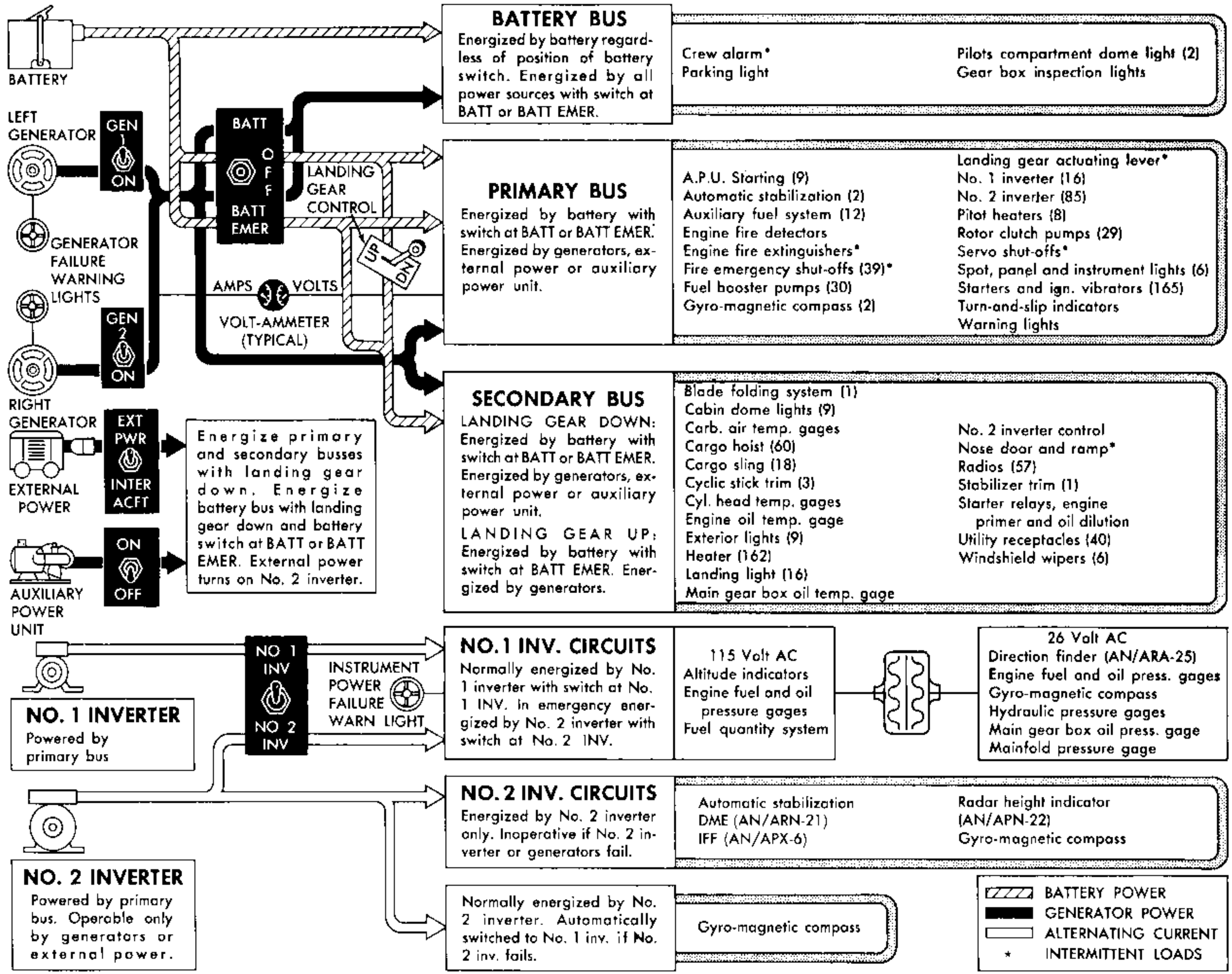


Figure 1—22. Electrical System Schematic Diagram

battery busses. The power sources are connected to the busses through automatic relays, the function of which is covered in the following paragraphs. All individual circuits for the operation of dc equipment are connected to the busses through circuit breakers.

GENERATORS.

A 28-volt, 300-ampere generator (13, figure 1—3) is mounted on and driven by the accessory section of each engine. The generators are connected in parallel. After the engines are started and operating at approximately 1300 rpm, the generator switches are turned on and the generators supply power for the operation of all electrical equipment. Normally, both generators are used, but the system is designed so that one generator will supply sufficient power for the operation of all equipment. When operating on one generator, all nonessential equipment should be turned off to prevent overloading the remaining generator. The output of each generator is controlled by a voltage regulator and a reverse current cut-out. The voltage regulator maintains constant generator output voltage regardless of engine speed and electrical load requirements by automatically controlling the generator field current. The reverse current cut-out prevents the battery from discharging through the generator whenever the generator is not running or its voltage falls below that of the battery. An equalizer circuit connects the two voltage regulators to equalize the load carried by each generator.

BATTERY.

The 24-volt, 20-ampere hour battery (10, figure 1—3) is located below the cabin floor aft of the cargo hatch. Battery power is used for limited ground operations, including emergency starting of the engine when no external power source or auxiliary power unit is available, and as an emergency source of power in the event of failure of both generators during flight.

EXTERNAL POWER RECEPTACLE.

The 28-volt external power receptacle (12, figure 1—3) is located on the right side of the fuselage forward of the cabin door and is accessible from outside the helicopter through an access door, marked EXTERNAL POWER 28 VOLTS DC. To prevent draining the battery, external power should be used for all ground operation up to the time the generators are turned on and reach operating speed (approximately 1300 engine rpm).

AUXILIARY POWER UNIT.

The 28-volt, 175-ampere auxiliary power unit (16, figure 1—3 and figure 4—12) is located in the left rear of the cabin. The auxiliary power unit may be used for operating the cargo hoist, testing of equipment on the ground, and for starting the engines when no external power source is available. For further description and operating procedures, refer to **AUXILIARY POWER UNIT,** Section IV. The automatic control relays, which connect the power sources to the busses, are contained in the dc power relay junction box, located in the cabin ceiling forward of the cargo door.

PRIMARY BUS.

The primary bus (figure 1—22) supplies power for the operation of the dc flight instruments and equipment most essential for safety of flight. The primary bus may be energized from all four power sources. When external power is plugged in, or when the auxiliary power unit is operating and turned on, it connects directly to the primary bus. Battery power is supplied to the primary bus whenever the battery switch is placed in either the BATT or the BATT EMER position. Generator power is supplied to the primary bus when a generator switch is placed in the ON position and the corresponding generator has reached operating speed. Should both generators fail during flight, battery power will be automatically distributed to the primary bus if the battery switch is on. The APU may be started to aid the battery in energizing the primary bus.

SECONDARY BUS.

The secondary bus (figure 1—22) supplies power for auxiliary equipment and equipment less essential for safety of flight. The secondary bus is energized by the primary bus when one or both of two generator secondary bus relays are closed. The left generator secondary bus relay is closed only when the left generator switch is on and the left generator is at operating speed. The right generator secondary bus relay is closed under any of three circumstances:

1. When the right generator switch is on and the right generator is at operating speed.
2. When the landing gear is down and the primary bus is energized.
3. When the battery switch is at BATT EMER.

The purpose of this arrangement is to permit the nonessential equipment connected to the secondary bus to be operated by all power sources while the helicopter is on the ground, but only on power supplied by one or both generators when the helicopter is airborne and the landing gear retracted. In case of failure of both generators during flight, the secondary bus may be energized by power from the battery through the primary bus when the battery switch is placed in the BATT EMER position. Under this condition, all nonessential equipment should be turned off to prolong battery life. The APU may be started to energize the primary bus. The secondary bus will also be energized by the battery when the landing gear is lowered, preparatory to landing with the battery switch in either the BATT or BATT EMER position.

Figure 1—23. External Power Receptacle and Switch

BATTERY BUS.

The battery bus (figure 1—22) supplies power to the gear box oil level inspection lights, the parking light, the pilot's compartment dome light, and the crew alarm bell, and is continuously energized regardless of the position of the battery switch.

GENERATOR SWITCHES.

Two guarded generator switches (figure 1—10), marked GEN 1 and GEN 2, located on the overhead switch panel, control the operation of the two generators. The switch, marked GEN 1, is for the generator driven by the left engine, and the switch, marked GEN 2, is for the generator driven by the right engine. Each generator supplies electrical power to the primary bus when its switch is placed in the ON position. The secondary bus receives power from the primary bus through a generator secondary bus relay for each generator. The electrical system is designed to operate on one generator, but when operating with one generator, all nonessential equipment should be turned off. Should both generators fail in flight (landing gear retracted), both secondary bus relays will open, disconnecting the secondary bus from the power supply system to minimize the load on the battery. All of the secondary bus relays are contained in the direct current relay junction box.

BATTERY SWITCH.

The battery switch (figure 1—10), marked BATT, OFF, and BATT EMER, is located between the two generator switches on the overhead switch panel. The BATT position closes the circuit through the main junction box to the primary bus. The secondary bus receives no power from the battery when the switch is in the BATT position except when the landing gear lever is in the DOWN position. Control current will then flow from the battery through the landing gear switch to close the right generator secondary bus relay which permits current to flow from the primary to the secondary bus. The OFF position disconnects the battery from the primary and secondary bus but does not disconnect the battery bus. The BATT EMER position energizes the primary bus and also closes the right generator secondary bus relay energizing the secondary bus. This position is used in event of failure of both generators to permit equipment operating from the secondary bus to be used in flight, if necessary, when the landing gear is retracted.

EXTERNAL POWER SWITCH.

A spring-loaded external power switch (figure 1—23), marked EXT PWR and INTER ACFT, is accessible from outside the helicopter through the external power receptacle hinged access door. The switch is normally in the EXT PWR position, and external power plugged in at the receptacle will be distributed to all busses. When the switch is placed in the INTER ACFT position, power from the primary bus closes the external power relay, permitting the external power receptacle to be used as a source of power for starting other aircraft or operating ground equipment. When an inter-aircraft external power cable is plugged in, it is only necessary to hold the switch in the INTER ACFT position momentarily to close the external power relay. Once the relay is closed, it will be held closed automatically by a jumper wire within the plug of the external power cable. The battery, auxiliary power unit, or one or both engines and generators must be operating to energize the primary bus before the external power receptacle can be used as a power source.

DIRECT CURRENT CIRCUIT BREAKERS.

Circuit breakers (figure 1—24), protecting the various dc circuits, are located on four panels in various parts of the helicopter. The battery bus circuit breaker panel is located in the cabin ceiling inboard of the dc power junction box. The two engine ignition vibrator circuit breakers (primary bus) are also located on this panel. The overhead circuit breaker panel is located behind the overhead switch panel in the pilot's compartment. The three rows of circuit breakers on the forward part of this panel are for equipment operating from the primary bus and the three rows on the aft part of the panel are for equipment operating from the secondary bus. The hoist and the ventilating fan circuit breakers (secondary bus) and the No. 2 inverter circuit breaker (primary bus)

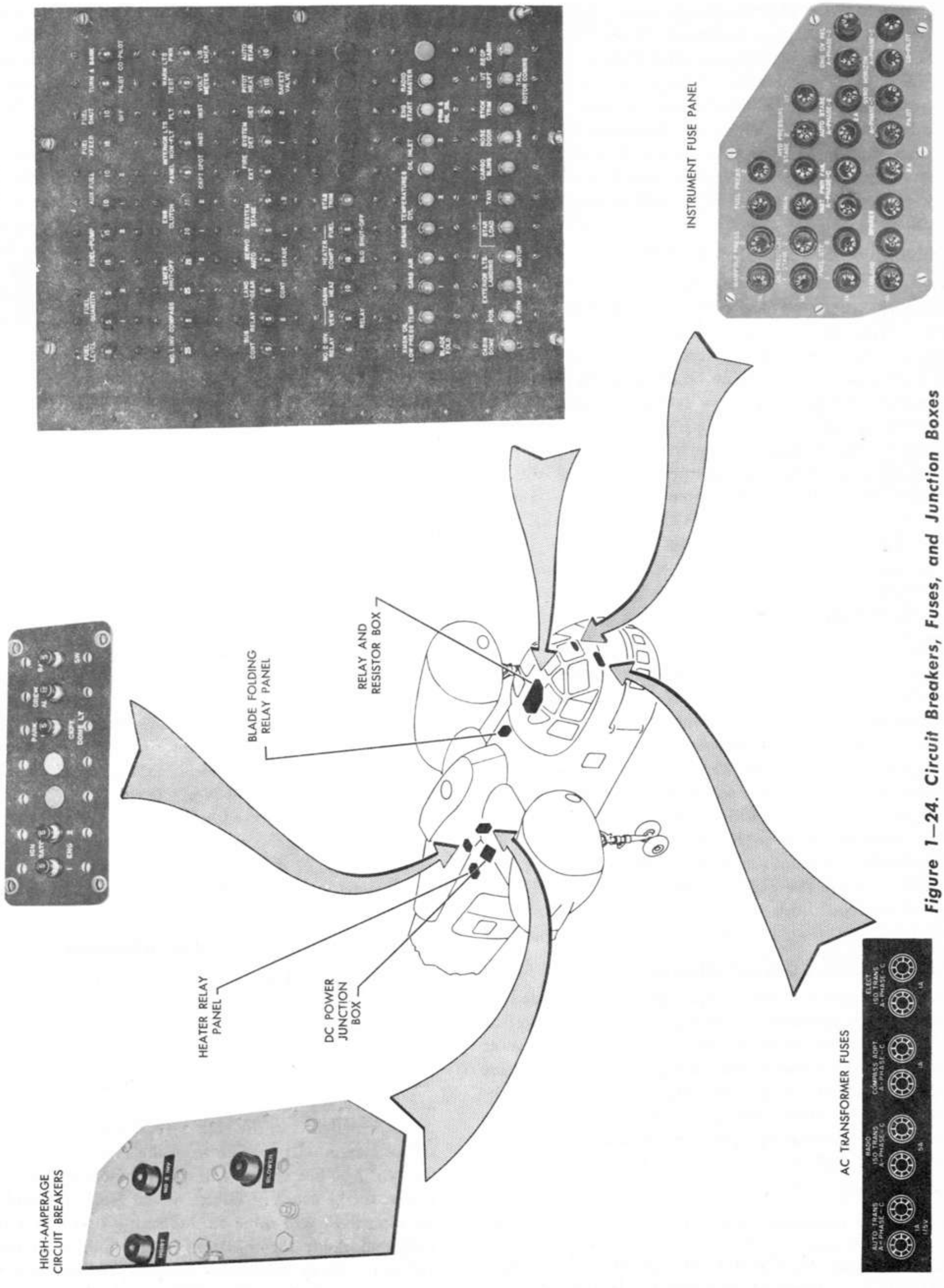


Figure 1—24. Circuit Breakers, Fuses, and Junction Boxes

are located on a high amperage circuit breaker panel forward of the battery bus panel in the cabin ceiling. The radio circuit breakers (secondary bus) are located on the radio circuit breaker panel on the aft face of the control console in the pilot's compartment. All circuit breakers are marked as to the operating circuit they protect and are of the push-pull type which may be reset. Any malfunctioning circuit may be isolated from the dc power supply system by pulling out its circuit breaker.

GENERATOR WARNING LIGHTS.

Two red generator warning lights (figure 1–10), one for each generator, are located above the generator switches on the overhead switch panel. The lights are of the press-to-test type and are marked GEN WARN. If a generator or an engine becomes inoperative, or if generator voltage falls below battery voltage, the corresponding generator warning light will go on. If both lights are on, it indicates that the primary bus is energized by other than generator power, and that the secondary bus is inoperative unless the landing gear is down or the battery switch is in the BATT EMER position. Should both warning lights go on in flight, all nonessential equipment should be turned off to minimize the load on the battery, and the APU should be started to supply power to the primary bus.

VOLT-AMMETERS.

Two volt-ammeters (40 and 46, figure 1–12) are located on the instrument panel. The volt-ammeter consists of a voltmeter and an ammeter mounted in a single case. The voltmeters indicate the voltage of the primary bus and should read between 22 and 24 volts when battery power is being used, and approximately 28.5 volts when the generator or the auxiliary power unit is being used. The ammeters indicate the load being placed on the generators. The pilot's ammeter indicates the right generator load and the copilot's ammeter indicates the left generator load. Both ammeters should read alike, with the loads increasing as more dc equipment is turned on.

ALTERNATING CURRENT POWER SUPPLY SYSTEM.

Power for the ac power supply system is provided by two 115-volt inverters (5, figure 1–3), the operation of which is controlled by the flight instrument power switch. Power is distributed through two ac bus systems which connect through the flight instrument power switch on the instrument panel. All ac operating circuits are protected by appropriately marked fuses on three fuse panels.

INVERTERS.

Alternating current is supplied by the No. 1 inverter rated at 115-volts ac, 250 va, 3-phase, 400 cps and the No. 2 inverter rated at 115-volts ac, 1500 va, 3-phase, 400 cps. The inverters are located under the forward section of the main rotor fairing. The No. 1 inverter operates directly from the primary bus through the NO. 1 INV circuit breaker on the overhead circuit breaker panel. The No. 2 inverter operates from the primary bus through the NO. 2 INV RELAY on the overhead circuit breaker panel, but the starting switch is energized by the secondary bus through the right and left generator bus control relays. The No. 2 inverter operates only when one or both generators are operating, or when external power is supplied. This arrangement prevents overloading the battery with the No. 2 inverter when no generator or external power is connected. The No. 1 inverter normally supplies alternating current for the instruments, and the No. 2 inverter supplies alternating current for the operation of the radio equipment.

ALTERNATING CURRENT DISTRIBUTION.

Power for the operation of the ac electrical equipment is distributed from the two inverters through two separate bus systems (figure 1–22). The No. 1 inverter bus system supplies power at 115 volts for the operation of the attitude indicators and the fuel quantity indicating systems, and at 26 volts for the pressure gages, the gyro-magnetic compass system, the UHF direction finder, and the radio-magnetic indicators. Transformers are used to reduce the voltage from 115 to 26 volts. The No. 2 inverter bus system supplies power at 115 volts for operation of the radar height indicator, the IFF equipment, the automatic stabilization equipment, and the course indicator. The No. 2 inverter also supplies power through a power adapter at 200 volts for the gyro-magnetic compass system. In the event of failure of the No. 1 inverter, all No. 1 circuits can be switched to the No. 2 inverter which will then be carrying the entire ac load. In the event of failure of the No. 2 inverter, all No. 2 circuits will be inoperative except the gyro-magnetic compass which will be switched automatically to the No. 1 inverter.

FLIGHT INSTRUMENT POWER SWITCH AND WARNING LIGHT.

The flight instrument power switch (24, figure 1–12), marked FLT INST PWR with positions NO. 1 INV and NO. 2 INV, and the instrument power failure warning light (25, figure 1–12), marked POWER FAILURE, are located on the instrument panel. When the two-position switch is in the NO. 1 INV position, the No. 1 ac bus system will operate from the No. 1 inverter, and the No. 2 ac bus system will operate from the No. 2 inverter. In the event of failure of the No. 1 inverter, the warning light will come on, indicating that power is no longer available for operation of the flight instruments, except for the gyro-magnetic compass. The switch is then placed in the NO. 2 INV position and all ac equipment will receive power from the No. 2 inverter and the warning light will go out. If the light then comes on, it indicates that the power to operate the flight instruments is lost, due to failure of the No. 2 inverter.

ALTERNATING CURRENT FUSES.

Fuses (figure 1–24) protecting the ac equipment operating circuits are located on various panels. The fuses in the flight instrument circuits are located on a panel on the left side of the control console. The fuses in the radio circuits are located on the radio fuse panel on the right side of the control console. Fuses in the transformer circuits are on the transformer fuse panel, located in the cabin forward of the pilot's compartment. All fuses are marked as to the circuits they protect. Spare fuses are provided on the radio and instrument fuse panels and on the structure adjacent to the transformer fuse panel. Spare fuses are also located on the forward portion of the control console in the pilot's compartment.

HYDRAULIC POWER SUPPLY SYSTEMS.

Four separate hydraulic power supply systems are installed in the helicopter: the first stage flight control servo system, the second stage flight control servo system, the utility system, and the emergency system. The first and second stage flight control servos operate from independent, self-contained hydraulic systems described under their respective headings in this section. The utility and emergency systems are covered in the following paragraphs.

UTILITY HYDRAULIC POWER SUPPLY SYSTEM.

Pressure for the utility hydraulic power supply system (figure 1–25) is supplied by a 3000 psi hydraulic pump mounted on, and driven by, the left engine. The reservoir (4, figure 1–45) for this system has a capacity of approximately 3.6 gallons plus expansion space, and is located in the forward part of the main rotor fairing. The reservoir filler and sight gage are accessible through doors at the top of the left fairing. Hydraulic fluid in the supply line leading from the reservoir to the pump on the left engine is cut off by an electrically operated engine fire emergency valve at the fire wall when the left-hand engine fire emergency handle on the overhead switch panel is pulled. Hydraulic power is distributed to the various actuating cylinders by means of a 3000 psi, a 1500 psi, and a 1000 psi supply line. An external source of hydraulic power for ground operation may be connected to the utility hydraulic system through three receptacles, located on the inboard side of the left main landing gear wheel well. The specification for the hydraulic fluid used in this system is shown on figure 1–45.

UTILITY HYDRAULIC PRESSURE GAGE.

Utility hydraulic system pressure is indicated in pounds per square inch on a gage (42, figure 1–12), located at the bottom center of the instrument panel. The gage operates on current from the No. 1 inverter, 26-volt alternating current bus, and is protected by a fuse, marked HYD PRESSURE UT, located on the instrument fuse panel.

EMERGENCY HYDRAULIC SYSTEM.

An emergency hydraulic system (figure 1–25) is provided to lower the landing gear only. The emergency system consists of a reservoir located in the cabin ceiling, which is automatically filled from the utility hydraulic reservoir, and a manually operated hydraulic pump and valve located to the left of the copilot's collective pitch control.

EMERGENCY HYDRAULIC SHUT-OFF VALVE.

A manually operated hydraulic shut-off valve (1, figure 1–8) is provided to the left of the copilot's collective pitch control. The toggle-type valve is normally open with the handle in a vertical position. The handle must be pushed to the horizontal position, snapping the safety wire, to close the valve before operating the emergency pump.

EMERGENCY HYDRAULIC PUMP LEVER.

The emergency hydraulic pump (7, figure 1–8) is located to the left of the copilot's seat. The pump has a telescoping lever which is extended forward and actuated manually to produce emergency hydraulic pressure.

EMERGENCY HYDRAULIC PRESSURE GAGE.

Emergency hydraulic system pressure is indicated on a gage (4, figure 1–8), located at the copilot's left, outboard of the emergency hydraulic pump handle. The gage is connected directly to the emergency system tubing, and functions only when the emergency hydraulic system is in operation. Do not actuate the emergency hydraulic pump lever to produce more than 1500 psi.

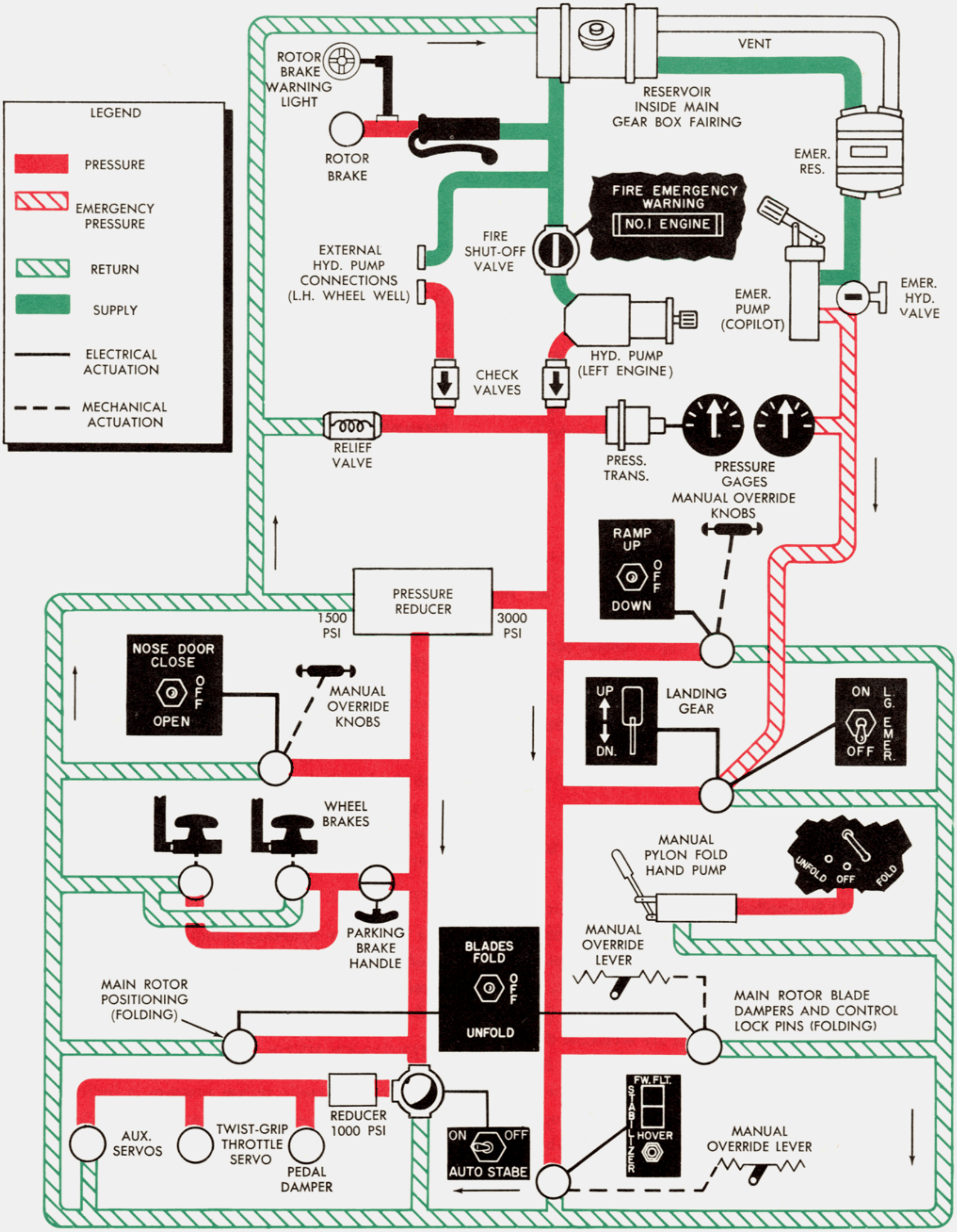


Figure 1—25. Utility Hydraulic System Schematic Diagram

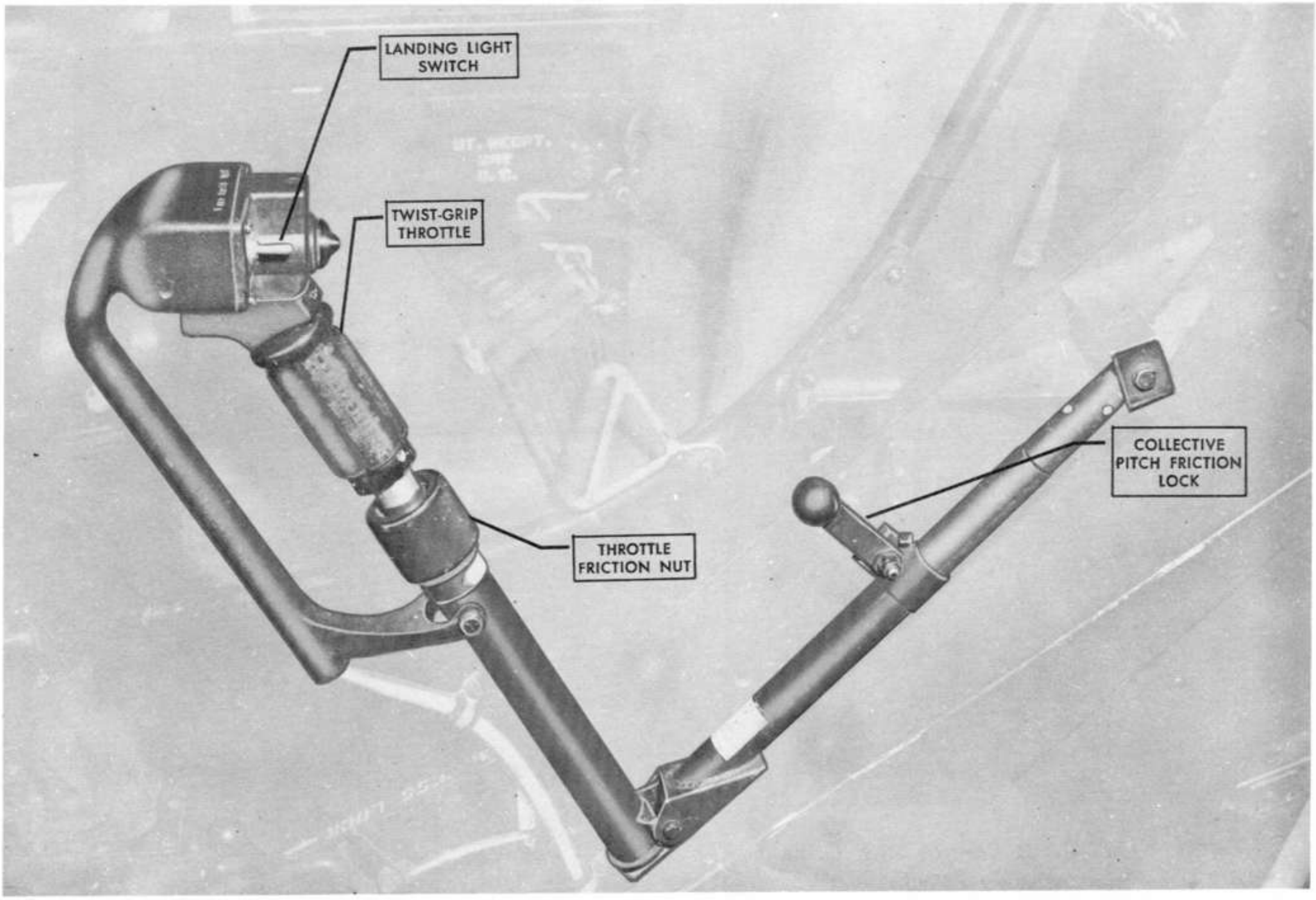


Figure 1—26. Pilot's Collective Pitch Control

FLIGHT CONTROL SYSTEM.

For descriptive purposes, the flight control system will be divided into several systems as follows: the main rotor flight control system, the cyclic control stick trim system, the tail rotor flight control system, the flight control servo systems, and the stabilizer control system.

MAIN ROTOR FLIGHT CONTROL SYSTEM.

The main rotor flight control system provides both vertical control and directional control. Vertical control is accomplished by changing the collective pitch of the main rotor blades to increase or decrease the angle of attack and consequently the lift developed by the blades. Directional control is accomplished by varying the pitch of each blade individually as it rotates, thus causing it to rise and fall cyclically and in effect tilting the tip-path plane of the main rotor blades to obtain a horizontal as well as a vertical component of thrust. The horizontal component of thrust will move the helicopter horizontally in whichever direction the plane of rotation is tilted. Control stick movements are transmitted from the pilot's compartment to the main rotor servo units through mechanical linkage. Control action is accomplished by hydraulically operated flight control servos. The control system is so designed that a change in either collective pitch for vertical control or cyclic pitch for directional control will not alter previous adjustment of the other. The main rotor flight controls terminate at the lower stationary star of the main rotor head. Control action is transmitted through the rotating star and linkage on the rotor head to the main rotor blades.

COLLECTIVE PITCH CONTROLS.

Two collective pitch controls (figure 1—26) are located in the pilot's compartment, one to the left of the pilot's seat, and the other to the left of the copilot's seat. Both controls operate simultaneously to change the collective pitch of the main rotor blades. From the minimum pitch position, the controls travel upward 10½ inches to the maximum pitch position. A friction lock on a telescoping rod, attached to the pilot's collective pitch control, may be pulled aft to apply friction to stiffen the control action or to prevent creeping while in flight. The rotatable handgrip of each collective pitch control is the twist-grip throttle which is partially synchronized to the vertical movement of the collective pitch control as

Figure 1—27. Cyclic Control Stick Grips (Prior to BuNo 145885)

described under TWIST-GRIP THROTTLES in this section. A switch box, located on the forward end of the pilot's collective pitch control, houses the landing light switches (figure 4—11). On helicopters with wide chord blades (S1515-20701), a pip pin stored in the bracket on the adjacent seat support is used to lock the collective pitch control for blade folding.

CYCLIC CONTROL STICKS.

A cyclic control stick (2, figure 1—5, and 6, figure 1—6), located in front of each seat in the pilot's compartment, provides the means of directionally controlling the helicopter. Moving the control stick in any direction tilts the tip-path plane of the main rotor blades in that direction and moves the helicopter in the same direction. The hand-grip of each cyclic control stick contains a two-position microphone trigger switch connected to the interphone system and to the radio transmitters. A switch box on the upper end of each cyclic control stick grip (figures 1—27 and 1—28) contains three thumb-operated switches. These switches control the stick trim system, the cargo release, and automatic stabilization disengagement. On BuNos 145855 and subsequent, the cyclic control stick grip contains switches which control the beeper trim system, the cargo release, and the automatic stabilization disengagement.

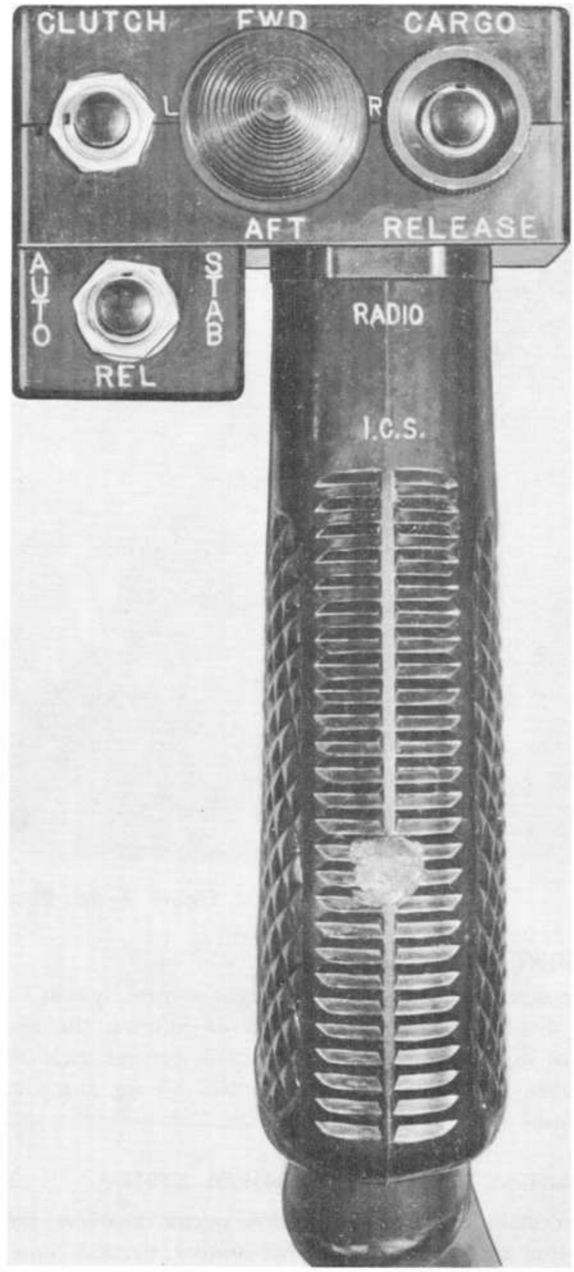


Figure 1—28. Cyclic Control Stick Grips (BuNos 145855 and Subsequent)

CYCLIC CONTROL STICK TRIM SYSTEM (PRIOR TO BUNO 145855).

The cyclic control stick trim system provides a method of positioning the cyclic stick for a desired helicopter attitude, and a means of holding the stick when the helicopter is stabilized by the ASE. The system consists of two springs enclosed in a cylinder, solenoid actuated magnetic brakes, control switches, and connecting linkage to the cyclic control stick. When the magnetic brakes are engaged, they hold the springs in a fixed position.

The springs then create a resistance of cyclic control stick movement. The springs exert a force of two pounds to hold the cyclic control stick in position. This force will increase proportionally to the distance the cyclic control stick is displaced from this position. When the magnetic brakes are disengaged, the springs are free to follow the cyclic control stick, offering no resistance to the cyclic control stick movement. Upon re-engagement of the magnetic brakes, the springs will again exert an initial force of two pounds regardless of where the cyclic control stick has been positioned. The trim feature is brought about by the ability to relocate the initial force of the springs.

CYCLIC CONTROL STICK TRIM MASTER SWITCH.

A cyclic control stick trim master switch, marked STICK TRIM (figure 1—10), is located on the overhead switch panel. When the switch is placed in the center (off) position, the magnetic brakes are disengaged by a solenoid and current is no longer supplied to the cyclic control stick trim thumb switches. If the cyclic stick is released with the master switch off, the cyclic stick will "fall away" and hit the stop. When the switch is placed in the ON position, the solenoid is de-energized, the magnetic brakes are engaged, and the current is supplied to the cyclic control stick trim thumb switches. The cyclic stick trim circuit operates on current from the secondary bus and is protected by a circuit breaker, marked STICK TRIM, located on the overhead circuit breaker panel.

CYCLIC CONTROL STICK TRIM RELEASE SWITCHES. A cyclic control stick trim release switch (figure 1—27), marked STICK TRIM, located on both cyclic control stick grips, enables the pilot or copilot to trim the cyclic control stick to any desired position. The switches are inoperative unless the master switch is placed in the ON position. Depressing the thumb switches disengages the magnetic brakes electrically. When the thumb switches are depressed, the magnetic brakes are released instantaneously and almost no force is required to move the cyclic control stick. If the thumb switch is depressed while the stick is out of the trimmed position, the stick will "jump" because of the almost instantaneous reduction in the force opposing the force exerted on the cyclic control stick by the pilot. Cyclic control stick "jump" can be avoided by depressing the thumb switch before the stick is moved from trim, and releasing it after the new position is established. When the switches are released, the solenoid is de-energized and the magnetic brakes are engaged. To trim the stick position, depress the thumb switch, move the cyclic control stick to the desired position, and then release the thumb switch. Stick trim will then be centered about the desired stick position.

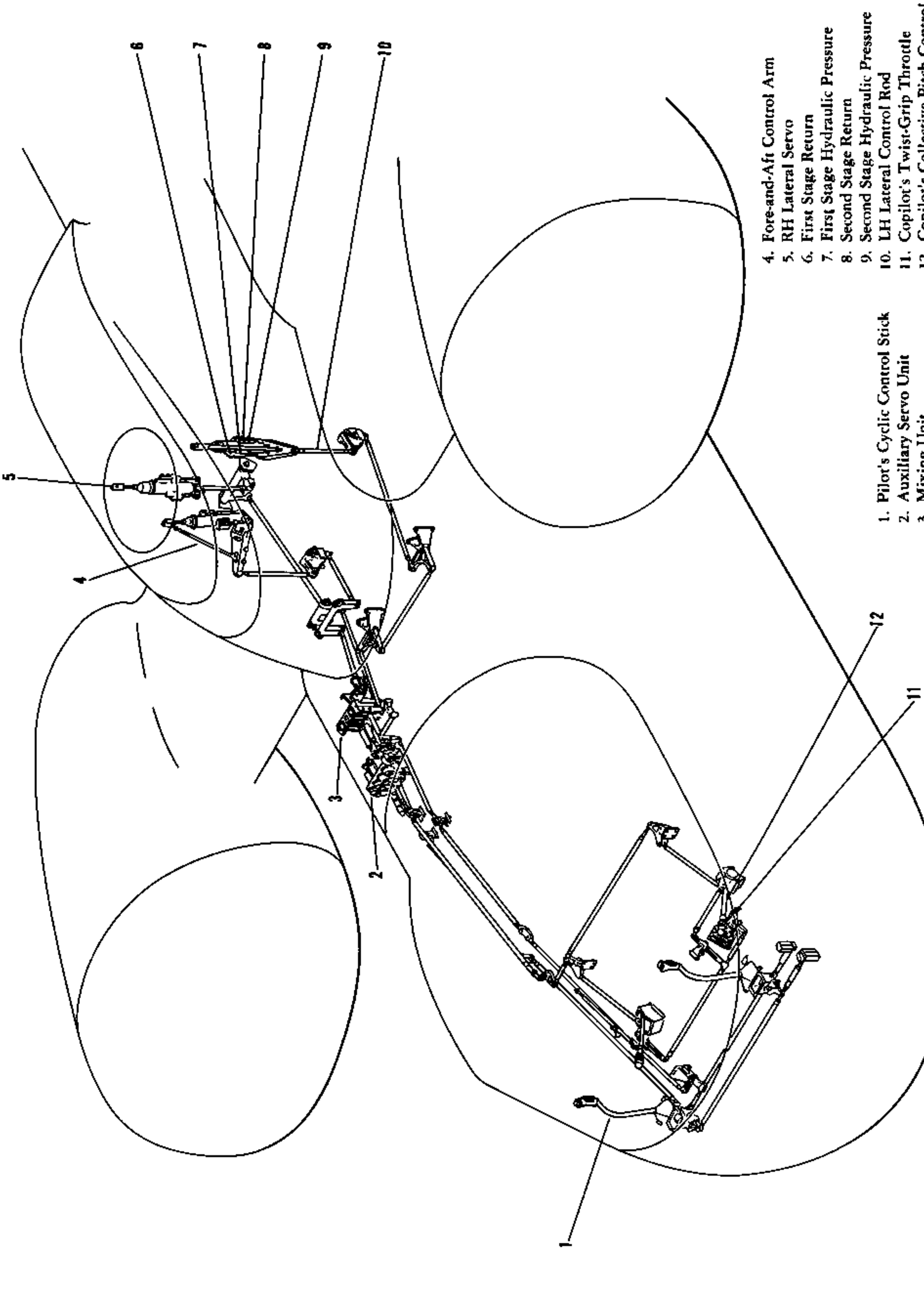
FORCE GRADIENT – BEEPER TRIM SYSTEM (BUNOS 145855 AND SUBSEQUENT).

The force gradient – beeper trim system provides the pilot with a precise means of trimming the helicopter's attitude. When used with the automatic stabilization equipment engaged, the system permits hands-off stability by holding the cyclic control stick in the desired position. Two actuators are powered by direct current from the secondary bus through a circuit breaker, marked BEEPER TRIM, located on the overhead breaker panel. The actuator output arm is connected to the cyclic control stick through a force gradient spring. There is one actuator and spring for fore-and-aft control and another actuator and spring for lateral control. The force gradient springs provide spring centering for the cyclic control stick when the actuator clutch is engaged. The clutch is engaged when the stick trim system is energized. When the clutch is disengaged, the output arm of the actuator is free, requiring only the force necessary to overcome the damper and the friction and inertia of the gear train. The actuator motor is energized by a four-position beeper switch, located on the pilot's and copilot's control stick grips. To trim the helicopter, the beeper switch is pushed in the desired direction and the actuators slowly move the cyclic control stick in that direction until the beeper switch is released. With the clutch engaged, the pilot may move the stick by overriding the force gradient spring. When the stick is released, it will return to its original position. The clutch may be disengaged by pressing a stick trim button, marked CLUTCH, located on the cyclic control stick grip. The stick can then be freely moved to any desired position. When the trim button is released, the cyclic control stick will be spring-centered in the new position. The force required at the stick to move the spring is approximately 1½ pounds to break out and ½ pound for each inch of cyclic control stick motion from the neutral position of the spring. Electrical power may be removed from the stick trim system by pulling the circuit breaker, marked BEEPER TRIM, located on the overhead circuit breaker panel, or by turning off the beeper trim master switch, located on the overhead switch panel.

BEEPER TRIM SWITCH. A four-way beeper trim switch (figure 1—28), marked FWD, AFT, L, and R, and spring-loaded to the center (off) position, is located on the pilot's and copilot's cyclic control stick grips. To trim the attitude of the helicopter, the beeper trim switch is pushed in the desired direction and the cyclic control stick will slowly move in that direction until the switch is released.

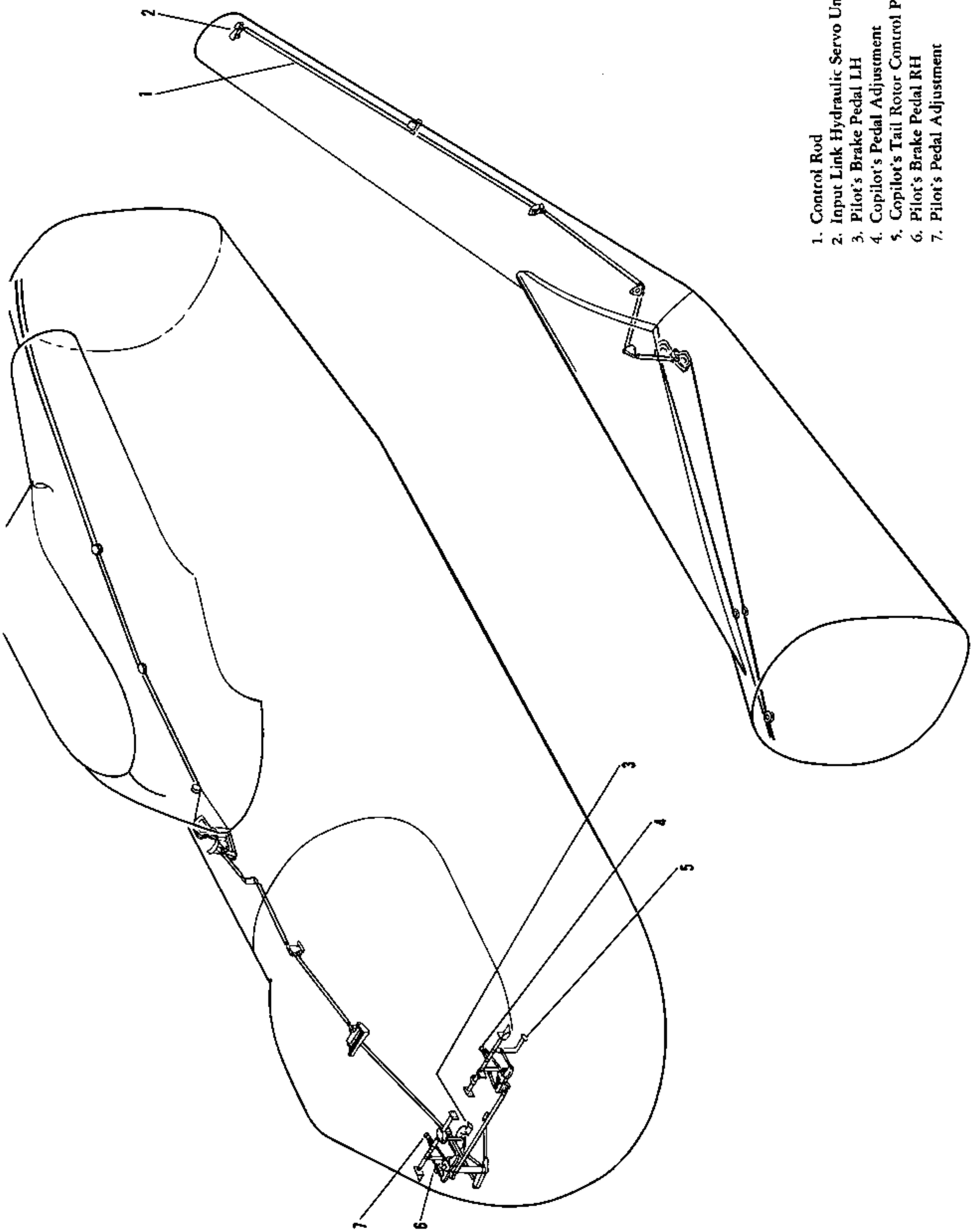
CYCLIC CONTROL STICK TRIM CLUTCH SWITCH.

A cyclic control stick trim clutch switch (figure 1—28), marked CLUTCH, is located on the pilot's and copilot's



1. Pilot's Cyclic Control Stick
2. Auxiliary Servo Unit
3. Mixing Unit
4. Fore-and-Aft Control Arm
5. RH Lateral Servo
6. First Stage Return
7. First Stage Hydraulic Pressure
8. Second Stage Return
9. Second Stage Hydraulic Pressure
10. LH Lateral Control Rod
11. Copilot's Twist-Grip Throttle
12. Copilot's Collective Pitch Control

Figure 1—29. Main Rotor Flight Controls System



1. Control Rod
2. Input Link Hydraulic Servo Unit
3. Pilot's Brake Pedal LH
4. Copilot's Pedal Adjustment
5. Copilot's Tail Rotor Control Pedal
6. Pilot's Brake Pedal RH
7. Pilot's Pedal Adjustment

Figure 1—30. Tail Rotor Flight Controls System

cyclic control stick grips. The button is pressed to disengage the actuator clutch, freeing the actuator output arm. The cyclic control stick can be freely moved to the desired position. When the button is released, the stick will remain spring-centered in the new position. This switch allows the pilot to easily move the cyclic control stick when maneuvering the helicopter to a new flight attitude.

BEEPER TRIM MASTER SWITCH. A beeper trim master switch, marked BEEPER TRIM, is located on the overhead switch panel (figure 1–10). When the switch is placed in the ON position, direct current flows from the secondary bus to the beeper trim system. If the switch is placed in the BEEPER TRIM (off) position, no current will flow from the secondary bus to the beeper trim system.

TAIL ROTOR FLIGHT CONTROL SYSTEM.

The functions of the tail rotor flight control system are to compensate for main rotor torque and to provide a means of changing the heading of the helicopter. The torque developed by the main rotor blades turning counterclockwise tends to rotate the fuselage in a clockwise direction. Gross weight, altitude, rate of climb, airspeed, and the corresponding power and collective pitch settings will vary the amount of main rotor torque. To compensate for torque variations, the pitch, and therefore the thrust of the tail rotor blades, can be increased or decreased collectively. Turns are accomplished by increasing tail rotor thrust, which overcompensates for main rotor torque and turns the helicopter to the left, or by decreasing the tail rotor thrust, which undercompensates for main rotor torque, and turns the helicopter to the right. Tail rotor control pedal movements are transmitted to the tail rotor servo unit assembly by mechanical linkage and cables. The tail rotor control cables are equipped with an automatic temperature compensator to maintain a constant cable tension under all climatic conditions. Control action is accomplished by hydraulically operated flight control servos. A tail rotor control pedal damper, connected to the utility hydraulic system, prevents abrupt movements of the tail rotor pedals and damage to the helicopter. The tail rotor control pedal damper is turned on and off simultaneously with the throttle servo and the automatic stabilization system servo by actuating the switch, marked AUTO STAB SERVO, located on the instrument panel.

TAIL ROTOR PEDALS AND ADJUSTMENT KNOBS.

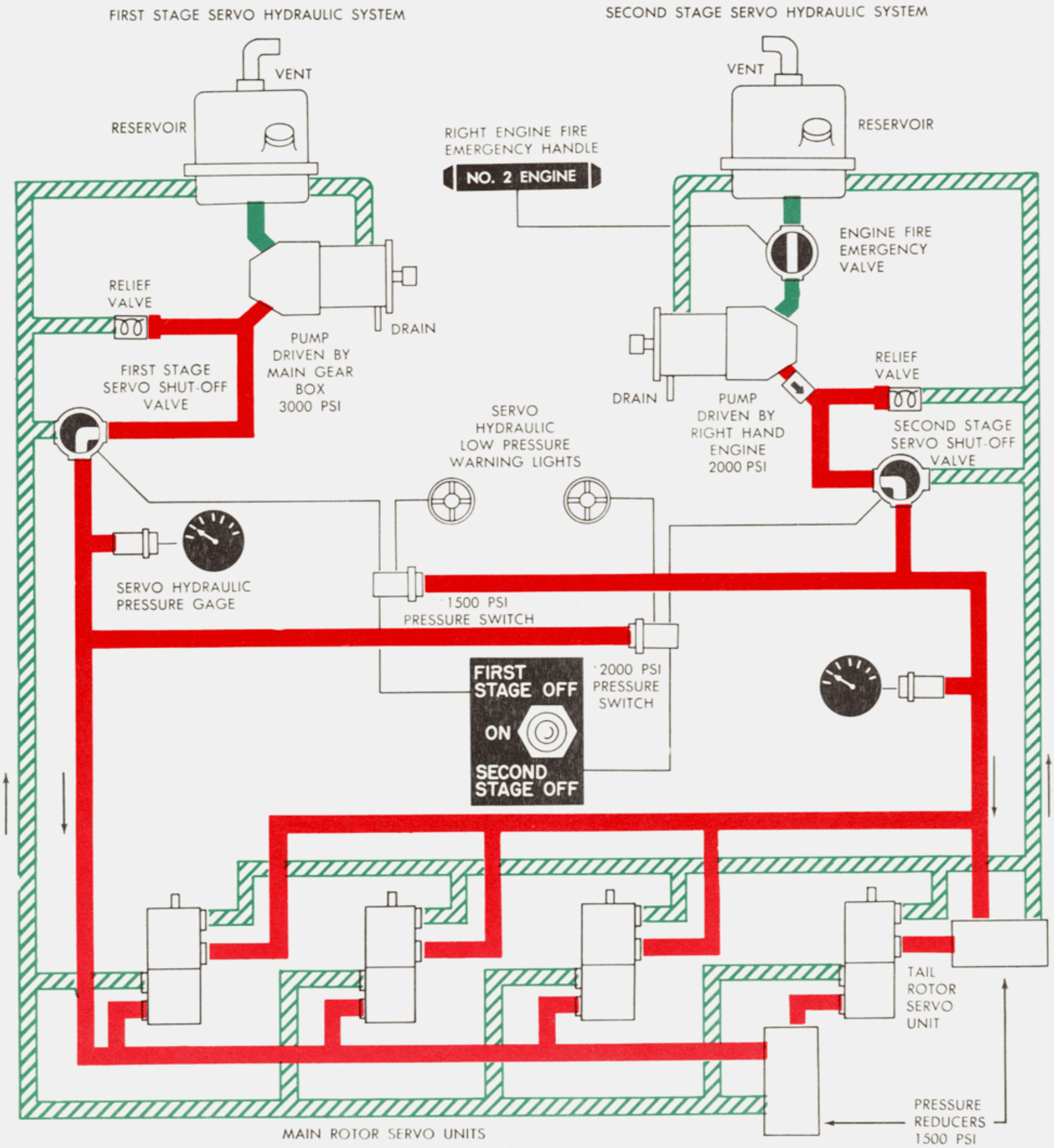
The tail rotor pedals (7, figure 1–6), one set in front of the pilot and the other in front of the copilot, change the pitch and thrust of the tail rotor, and, consequently, the heading of the helicopter. Tail rotor pedal adjustment knobs (5, figure 1–5, and 8, figure 1–6), that adjust the pedals for leg length, are located below the instrument panel. Toe brake pedals for the main landing gear wheel brakes are mounted on the pilot's tail rotor pedals.

FLIGHT CONTROL SERVO SYSTEM.

A flight control servo hydraulic system (figure 1–31) is incorporated into the flight control system. The servo system eliminates stick forces required by the pilot to operate the controls and those caused by main rotor vibratory loads.

The servo system utilizes four hydraulic piston units to vary the main and tail rotor blades pitch. Each servo unit consists of two independent stages connected to two completely independent hydraulic systems, each with its own pump, pressure control circuit, and reservoir. Only the piston rod and housing are common to the two stages of the unit. The hydraulic system for the first stage is powered by a pump driven by and mounted on the main gear box. The hydraulic system for the second stage is powered by a pump driven by and mounted on the right engine. With this arrangement, the pilot is assured of having the servo system operative not only prior to and during rotor head engagement, but also, in the event of engine failure, during autorotative landings. Three main servo units are mounted between the main gear box and the main rotor stationary star. A two-stage tail rotor servo unit is mounted on the tail gear box.

Each servo unit has two servo valves actuated by a common input. In operation, input signals, introduced by either the pilot or the automatic stabilization equipment, are transmitted through the control system to the input of the servo unit. This motion opens both of the servo valves, and directs hydraulic pressure simultaneously to both stages of the power piston, extending (or retracting) the piston. The power piston "output" is then directed to the stationary star linkage to provide the power to change main rotor blade pitch. The output motion of the power piston is connected back to the input causing the servo valves to return to their neutral position. Normally, both servo systems are in operation at all times. The flight control system also includes an auxiliary servo unit. The prime function of this unit is to serve as a means of introducing automatic stabilization equipment inputs to the flight control system. The auxiliary servo unit is powered from the utility hydraulic system pump which is driven by the left engine. The auxiliary servo unit is controlled by a switch, marked AUTO STAB SERVO, located on the instrument panel. The hydraulic fluid specification is shown in figure 1–45.



*IF HYDRAULIC PRESSURE IN FIRST STAGE SYSTEM DROPS TO 2000 PSI OR IF SYSTEM IS SHUT OFF, ITS PRESSURE SWITCH TURNS ON WARNING LIGHT AND PREVENTS REMAINING SYSTEM FROM BEING SHUT OFF.

**IF HYDRAULIC PRESSURE IN SECOND STAGE SYSTEM DROPS TO 1500 PSI OR IF SYSTEM IS SHUT OFF, ITS PRESSURE SWITCH TURNS ON WARNING LIGHT AND PREVENTS REMAINING SYSTEM FROM BEING SHUT OFF.

Figure 1—31. Flight Control Servo Hydraulic System Schematic Diagram

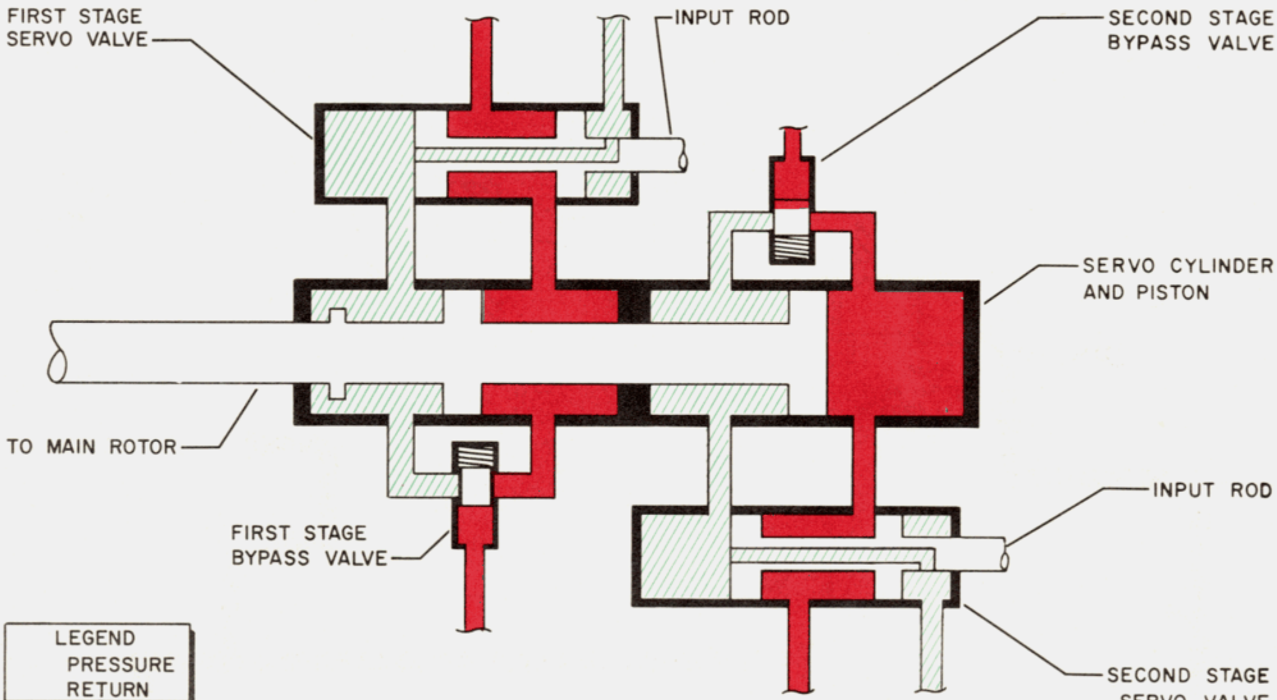


Figure 1—32. Main Rotor Servo Unit Schematic Diagram

FIRST STAGE SERVO HYDRAULIC SYSTEM.

The first stage servo hydraulic pump is driven by the accessory drive section of the main gear box so that it is in operation whenever the rotors are turning. The pump output of 3000 psi is utilized to power the main rotor first stage servo cylinders, but is reduced to 1500 psi for the tail rotor first stage servo cylinder. The first stage servo hydraulic reservoir (6, figure 1—45), mounted on the left side of the main gear box, has a capacity of approximately 0.4 gallon of hydraulic oil. A filler cap and an oil level sight gage are on the reservoir. The reservoir filler cap is accessible by removing the left-hand gear box fairing. The sight gage is visible through a window in the fairing.

SECOND STAGE SERVO HYDRAULIC SYSTEM.

The second stage servo hydraulic pump is driven by the accessory drive section of the right engine, so that if the right engine fails, the second stage servos are inoperative. The pump output of 2000 psi is utilized to power the main rotor second stage servo cylinders, but is reduced to 1500 psi for the tail rotor second stage servo cylinder. Hydraulic fluid in the supply line leading from the second stage servo hydraulic reservoir (3, figure 1—45) to the pump on the right engine is cut off at the fire wall by an electrically operated engine fire emergency valve when the right engine fire emergency handle on the overhead switch panel is pulled. The reservoir, mounted in the forward part of the main rotor fairing, has a capacity of approximately 0.7 gallon of hydraulic fluid. A filler cap and an oil level sight gage are on the reservoir. The filler cap is accessible through a door in the fairing. The sight gage is visible through a window in the fairing.

FIRST AND SECOND STAGE SERVO SHUT-OFF SWITCH AND WARNING LIGHTS.

Hydraulic pressure to either the first stage or the second stage servo cylinder is shut off by servo shut-off valves controlled by a three-position first and second stage servo shut-off switch (45, figure 1—12), marked SERVO SYS, and located on the center section of the instrument panel. The switch has three marked positions: ON, FIRST STAGE OFF, and SECOND STAGE OFF. Both servo hydraulic systems are normally in operation with the switch in the center ON position. To turn off the first stage servo system, the switch is placed in the FIRST STAGE OFF position; and to turn off the second stage servo system, the switch is placed in SECOND STAGE OFF position. However, the two systems are interconnected electrically with pressure switches in such a way that it is impossible to shut off the first stage unless there is a minimum of 1500 psi hydraulic pressure in the second stage, and it is impossible to shut off the second stage unless there is a minimum of 2000 psi hydraulic pressure in the first stage. Two red first and second stage servo hydraulic low pressure warning lights (45, figure 1—12), marked LOW PRESS, FIRST STAGE, SECOND STAGE, will light if the pressure is below its respective pressure switch setting. The servo shut-off valves and warning lights operate on direct current from the primary bus.

SERVO HYDRAULIC PRESSURE GAGES.

Two servo system hydraulic pressure gages (41 and 43, figure 1—12), located at the bottom of the instrument panel, operate on current from the No. 1 inverter 26-volt alternating current bus, and are protected by fuses, marked HYD PRESSURE 1 STAGE 2, located on the instrument fuse panel. The upper gage indicates the pressure in the first stage servo system; the lower gage indicates the pressure in the second stage servo system. Both gages are graduated in pounds per square inch.

AUTOMATIC STABILIZATION EQUIPMENT (ASE).

The automatic stabilization equipment (ASE) has been installed to provide the helicopter with flight characteristics similar to those of fixed-wing aircraft. While the cyclic stick is held in a fixed position, ASE will return the helicopter to a condition of equilibrium or steady flight from which the helicopter has been disturbed, damping out any oscillations encountered as a result of the disturbance. Thus, static and dynamic stability are achieved. ASE assists the pilot by counteracting the external forces tending to displace the helicopter from its flight path and causes the helicopter to respond in a stable manner to the maneuver called for by the flight controls. The equipment accomplishes this by adding or subtracting small limited authority corrections to the main rotor, without interfering with the normal relationship between the cyclic stick and the main rotor. In this respect, ASE is different from the autopilot used in fixed-wing aircraft in that it is engaged at all times and does not effect the normal use of the flight controls in any way. Thus, in pitch and roll, no override forces are present. In correcting for deviations in yaw and altitude, ASE may introduce visible movements in the tail rotor control pedals and collective pitch stick. These movements can be overcome by the pilot, if so desired, and, under the most severe circumstances, will present a maximum of 50 pounds force in the tail rotor control pedals and 10 pounds in the collective pitch stick.

The automatic stabilization equipment has two modes of operation: (1) attitude and directional stabilization and (2) altitude control. In the attitude mode of operation, the auxiliary servo unit holds pitch and roll angles of the fuselage constant by means of signals received from a vertical gyro reference, and the fuselage heading is held constant by signals received from the GE type MA-1 gyro compass system. Changes in pitch and roll angles and changes in heading made by the pilot will be held by the automatic stabilization equipment. Small pitch and roll stabilization corrections are

Figure 1—33. Automatic Stabilization Control Panel

constantly added even though the pilot moves or holds the cyclic control stick, but directional stabilization occurs only when the pilot's feet are removed from the tail rotor control pedals. Any one of four microswitches, one on each tail rotor control pedal, disengages the system when the pilot's feet are on the pedals. In the constant altitude mode of operation, altitude is stabilized through the auxiliary servo unit by signals received from a barometric altitude control. Only enough collective friction to suppress collective stick hunting in gusty weather should be used after engagement of this mode. Changes in altitude are accomplished by disengaging the altitude mode of automatic stabilization, actuating the flight controls normally to change altitude, and re-engaging the altitude mode after stabilizing the altitude, attitude, and airspeed of the helicopter. Since the fuselage pitching attitude determines airspeed, the first and second modes combine to provide automatic cruising flight with a constant airspeed and heading at a constant altitude.

Automatic stabilization will function with or without the stick trim system being engaged; however, the stick trim system should be used in conjunction with the automatic stabilization equipment to hold the stick at the desired flight condition and the stabilization equipment be allowed to correct for gusts, etc. during flight. If the stick trim system is not used and if the pilot attempts to correct for gusts, etc. by movements of the flight controls while automatic stabilization is engaged, some overcontrol and roughness may be encountered. Alternating current for the operation of the automatic stabilization components is supplied by the No. 2 inverter, and direct current is supplied by the primary bus. The auxiliary servo system provides the means of actuating the flight control system on signals received from the automatic stabilization equipment.

AUTOMATIC STABILIZATION CONTROL PANEL.

The automatic stabilization control panel (figure 1—33) is located on the control console on the pilot's side. The controls consist of a center-of-gravity trim control knob, a null indicator, and a yaw trim control knob extending

Figure 1—34 deleted.

across the top of the panel. Two engage buttons are at the bottom left of the panel.

The control knob, marked CG TRIM, is used to trim the automatic stabilization pitching control. The relation between actual cg location and pitching control trim may be checked in flight only by the dial and pointer, marked NULL INDIC. If this indicator is oscillating about an average position considerably off center during a steady hover, the cg trim is incorrectly set and may be adjusted by moving the cg control knob and readjusting cyclic stick position. The adjustment will assure adequate automatic stabilization equipment pitch control through all conditions of flight.

The control knob, marked YAW TRIM, permits the pilot to trim accurately the heading of the helicopter for cruise control with his feet off the tail rotor control pedals. The YAW TRIM knob is used for small changes only. Index marks are located on the panel around the knob and an index mark is on the knob. The knob must be pushed in and then turned to change heading. Rotating the knob from one index mark to the next will change the heading of the helicopter one degree. One revolution in either direction will produce a 10-degree change in heading. The position at which the knob is left is not important, since when the knob is released, it is disengaged from the system and may be rotated freely without producing a change in heading.

The engage buttons control the two modes of operation of the automatic stabilization equipment. The button on the left, marked ENGAGE, controls mode 1. The next button, marked BAR ALT, controls mode 2. When either of the engage buttons are pushed in, a green light in the center of the button will light signifying that the mode of operation controlled by that button is in operation. The automatic stabilization equipment is warming up as soon as the primary bus is energized and the No. 2 inverter is operating. To engage the system, push in the ENGAGE button about 3 minutes after

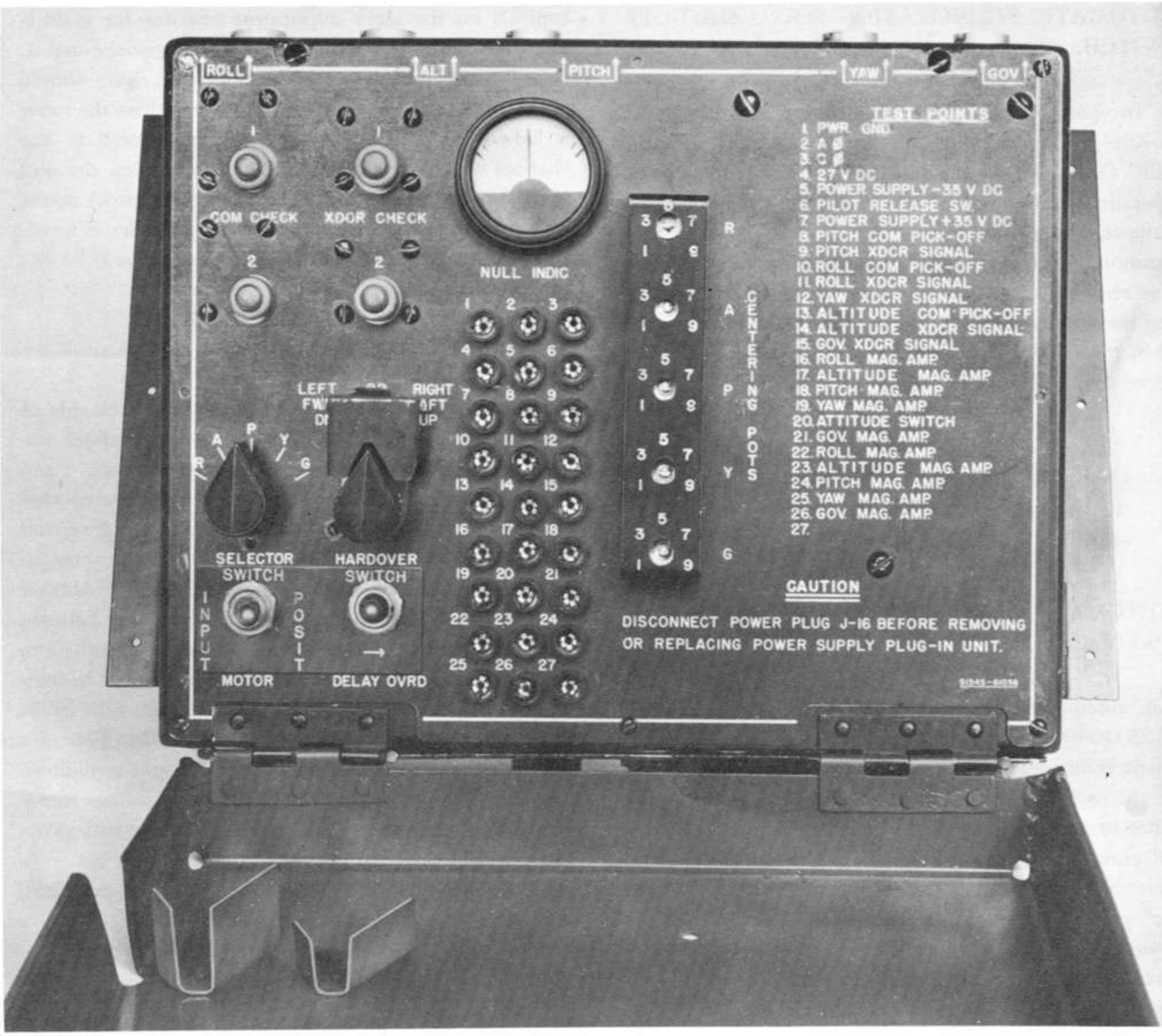


Figure 1—35. Automatic Stabilization Servo Adjustment Box (BuNos 140314 and Subsequent)

power is available and hold it in until the green light comes on. The ENGAGE button must be depressed and mode 1 operating before the other mode can be used.

AUTOMATIC STABILIZATION RELEASE SWITCH.

An automatic stabilization release switch (figure 1—28), marked AUTO STAB RELEASE, located on the grip of each cyclic control stick, is depressed to cut off all modes of automatic stabilization. Pressing the switch on either stick disengages the automatic stabilization equipment, and a two-second time delay relay electrically nulls the servo motors and then cuts off the power to all the electric servo motors. The motors are then held null mechanically. The automatic stabilization equipment remains warmed up, however, and may be re-engaged at any time by pushing the ENGAGE button on the automatic stabilization control panel. For normal disengagement, pulling out the ENGAGE button on the control panel will allow electric power to remain on the motors which is necessary for certain maintenance and hard-over checks. The automatic stabilization equipment should be disengaged after each flight while the No. 2 inverter is operating to make sure that the auxiliary servo pilot valves are not preset to an unknown position for next engine start. The disengagement may be made either by pulling out the ENGAGE button on the control panel or by pushing the automatic stabilization release switch on the cyclic control stick grip.

AUTOMATIC STABILIZATION SERVO SHUT-OFF SWITCH.

A two-position automatic stabilization servo shut-off switch (44, figure 1–12), marked AUTO STABE SERVO, located on the instrument panel, controls the operation of the auxiliary servo unit, the tail rotor pedal damper, and the twist-grip throttle servo. In the ON position, utility hydraulic pressure from the 1500 psi line reduced to 1000 psi supplies the power for operating these units. In the OFF position, the auxiliary servo unit, the tail rotor damper, and the twist-grip throttle servo are inoperative.

Note

Check that ASE is not engaged before placing the automatic stabilization servo shut-off switch in the OFF position.

AUTOMATIC STABILIZATION HARD-OVER SWITCH.

An automatic stabilization hard-over switch, marked HARDOVER, LEFT, RIGHT, is located on the automatic stabilization servo adjustment box (figure 1–35) in the main cabin. This switch is provided to enable the pilot to personally check the auxiliary servo pilot valve adjustment and the ability of the flight controls to override the automatic stabilization signals. This switch must be operated by a crew member in the cabin. When the switch is placed in either extreme position, the automatic stabilization equipment introduces steady, full-authority signals simultaneously on all four servo motors. With the motors in this condition, the cyclic control stick should be free to move full travel in all directions; the tail rotor control pedals will exert a limited force, but should be able to be restrained with a maximum force of 25 pounds and be free to move to their limits with a maximum force required of 50 pounds; the collective override force should not exceed 10 pounds. Further details for making this check are explained in section II. The switch must be in the OP (off) position except when accomplishing this check. A guard in the inside cover of the servo adjustment box will not allow the cover to close if either the hard-over or channel selector switches are not properly positioned. The hard-over switch should never be operated in flight as an automatic stabilization equipment full-authority hard-over signal will be applied to all channels even though the automatic stabilization is not engaged. All other controls on the servo adjustment box are for maintenance purposes and should otherwise be positioned as follows: the rotary switch, marked SEL SW, should point upward to the P (pitch) position before the cover is closed. This selector switch may be rotated to any channel which the pilot desires to monitor on the null indicator, but should always be reset to P (pitch) so the null indicator can be used with the cg trim knob to adjust cg trim in flight. All other switches should be left in their spring-loaded positions.

STABILIZER CONTROL SYSTEM.

The stabilizer (27, figure 1–2), located on each side of the aft fuselage, aids in maintaining longitudinal stability during forward flight under various loading and airspeed conditions. The stabilizer has a fixed horizontal position (forward flight) and a free swinging vertical position (hover). The stabilizer rotates about a torque tube that extends through the fuselage. Stabilizer damper struts (28, figure 1–2) are attached between the top of the stabilizer and the side of the fuselage to reduce vibration in the stabilizer. An associated hydraulic unit is mounted aft and below the torque tube inside the fuselage with an attachment to the tube. This unit serves to hold the stabilizer in a two-degree nose-down position (forward flight) and to dampen stabilizer movements in a vertical position (hover). A solenoid valve, energized by direct current from the secondary bus, controls the position of the stabilizer. When de-energized, the valve remains open, permitting hydraulic pressure to hold the stabilizer in a two-degree nose-down position. When energized, the solenoid valve closes, cutting off hydraulic pressure, and the stabilizer assumes a free swinging vertical position.

The large area of the stabilizer presented to the main rotor downwash creates excessive drag during hovering, tending to force the tail down. For best hovering performance, the stabilizer must be fully feathered with respect to the main rotor downwash. The stabilizer's effect on control and performance during hovering is negligible in this hydraulically dampened, free-swinging position. When the hydraulic pressure has been relieved, an amber light will come on indicating that the stabilizer is no longer being held in the forward flight position. This press-to-test light operates on direct current from the primary bus and is actuated by a pressure switch in the hydraulic unit. Prior to returning the stabilizer to the forward flight position, enough forward speed must be attained to force the stabilizer to an inter-

Figure 1–36 deleted.

mediate point between the forward flight and hover positions. When the stabilizer is at an intermediate point, enough hydraulic pressure is exerted on the pressure switch to turn the amber light off, and return the stabilizer to the forward flight position.

STABILIZER POSITIONING SWITCH.

The stabilizer positioning switch (figure 1–37), marked STABILIZER, with positions FWD FLIGHT and HOVER, is mounted on the right side of the control console in the pilot's compartment. When the switch guard is closed, the switch is in the FORWARD FLIGHT position and the solenoid valve in the hydraulic unit is de-energized. When the switch is placed in the HOVER position, the solenoid valve is energized. The circuit operates on direct current from the secondary bus, and is protected by a circuit breaker, marked STAB TRIM, located on the overhead circuit breaker panel.

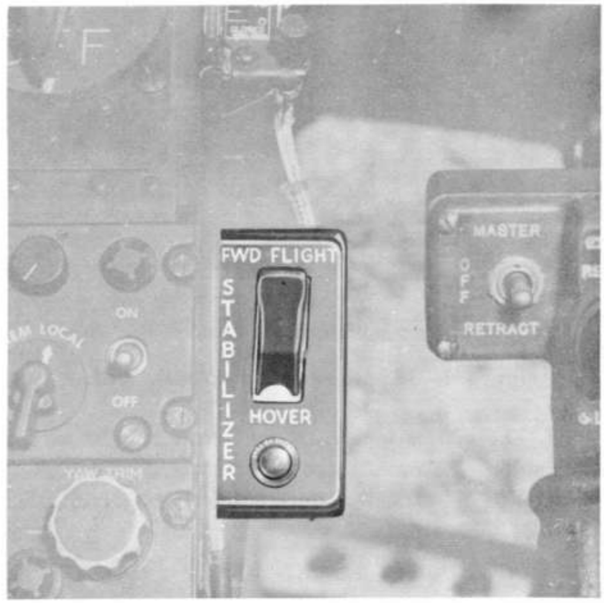


Figure 1–37. Stabilizer Positioning Switch and Warning Light (BuNos 140314 and Subsequent)

STABILIZER HOVERING LIGHT.

An amber, stabilizer hovering light (figure 1–37) is located aft of the stabilizer positioning switch. When hydraulic pressure is being applied to the stabilizer, holding it in the forward flight position, the light will remain out. When the hydraulic pressure is relieved, a pressure switch senses the drop in pressure and turns the light on.

STABILIZER HYDRAULIC VALVE.

The stabilizer hydraulic valve (figure 1–38) that controls stabilizer servo actuation is located in the aft fuselage section, in the upper right-hand corner, just inside the aft cabin bulkhead door. This valve is normally operated electrically. If electrical failure should occur when the stabilizer is in the forward flight position, and it is desired to place the stabilizer in the free-swinging hover position, it is necessary to push the valve manual override lever aft and hold it there as long as required. When the valve manual override lever is released (with no secondary bus electric power available at the valve) the stabilizer will return to the forward flight position whenever the helicopter develops sufficient forward

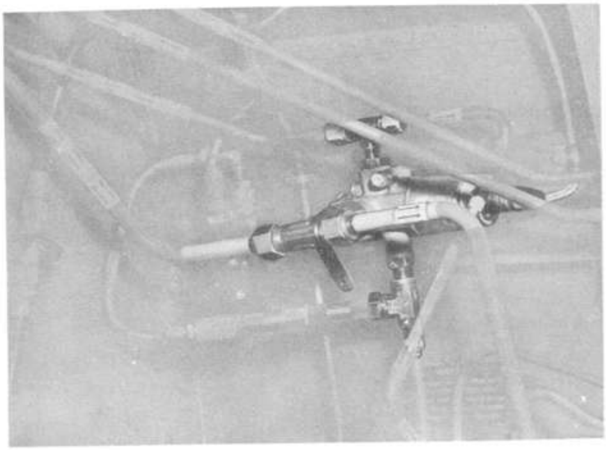


Figure 1—38. Stabilizer Hydraulic Valve

speed. The manual override lever must also be used if the stabilizer becomes locked in the overcenter position because of electrical failure. Refer to **STABILIZER FAILURE,** Section III.

STABILIZER GUST LOCK.

Helicopters modified in accordance with Aircraft Service Change No. 26 are provided with a stabilizer gust lock, which is installed to prevent buffeting damage to the stabilizer when the helicopter is moored. Eyebolt fittings are installed on the right rear fuselage and the right stabilizer. The gust lock consists of a steel rod which holds the stabilizer immobile, linking it to the right side of the fuselage, strut fashion.

LANDING GEAR SYSTEM.

The landing gear is of the conventional type consisting of two retractable main landing gear assemblies and a full-swiveling tail wheel.

MAIN LANDING GEAR. The two main landing gear assemblies (21, figure 1—2) are located below the engine nacelles and retract rearward and upward into the engine nacelles. The wide tread affords excellent lateral stability for landing and ground handling. The dual main landing gear wheels are equipped with tubeless tires and hydraulic brakes. The main landing gear is retracted and extended, and the up-lock and down-lock are released by power from the utility hydraulic system, controlled by direct current from the primary bus. (See figure 1—40.) The overall height of the helicopter may be decreased six inches by actuating a stowage valve lever on each landing gear which bleeds a quantity of hydraulic fluid from each shock strut.

TAIL WHEEL. The tail wheel (29, figure 1—2) is located in the conventional position underneath the aft section of the fuselage. The tail wheel is full-swiveling and self-centering. The tail wheel may be locked in the center position to prevent the aft end of the fuselage from swinging around under windy conditions on the ground or carrier decks.

LANDING GEAR ACTUATING LEVER.

A landing gear actuating lever (figure 1—40), marked LDG GEAR CONTROL, UP, DN, and located on the forward section of the control console, controls the retraction and extension of the main landing gear. The landing gear actuating lever operates a switch which utilizes direct current from the primary bus to operate the landing gear actuating solenoid valve in the utility hydraulic system. The lever with a knob in the shape of a miniature wheel is pushed forward to the UP position to retract the gear and is pulled aft to the DN position to extend the gear. The lever is locked in the DN position when the landing gear oleo shock struts are compressed, thus preventing the gear from being raised when the helicopter is on the ground. When the weight of the helicopter is not on the shock struts, the lever is automatically unlocked by a solenoid. When the lever is placed in the UP position, the retraction cycle starts by releasing the landing gear down-locks. A red warning light, located in the knob of the landing gear actuating lever, goes on and remains on whenever the main landing gear is out of the up and locked, or down and locked, position. The warning light may be tested during flight by pushing the button, marked WARNING LIGHT TEST, located on the panel to the left of the landing gear lever. The light cannot be tested when the helicopter is on the ground. If the secondary bus is de-energized in flight through failure of both generators, battery power automatically is switched to the secondary bus when the landing gear actuating lever is placed in the DN position and the battery switch is in the BATT position. The landing gear control circuit operates on direct current from the primary bus, and is protected by a circuit breaker, marked LAND GEAR CONT, located on the overhead circuit breaker panel.

LANDING GEAR STOWAGE VALVE LEVER.

A manually operated stowage valve, located on the outboard side of each main landing gear shock strut, makes it possible to decrease the overall height of the helicopter approximately six inches. When the valve lever is pulled down, a quantity of hydraulic fluid is displaced from the shock strut into the drag strut, permitting the weight of the helicopter to compress the shock strut. After the shock strut has been compressed, the lever should be released and will automatically return to the up position. At take-off, the weight of the wheels extends the strut and the displaced fluid automatically flows back into the shock strut. At the next landing, the heli-

1. Emergency Bypass Valve
2. Landing Gear Actuating Solenoid Valve
3. Utility Hydraulic Supply Reservoir
4. Hydraulic Lines
5. Landing Gear Actuating Cylinder LH
6. Landing Gear Up-Lock
7. Landing Gear Down-Lock

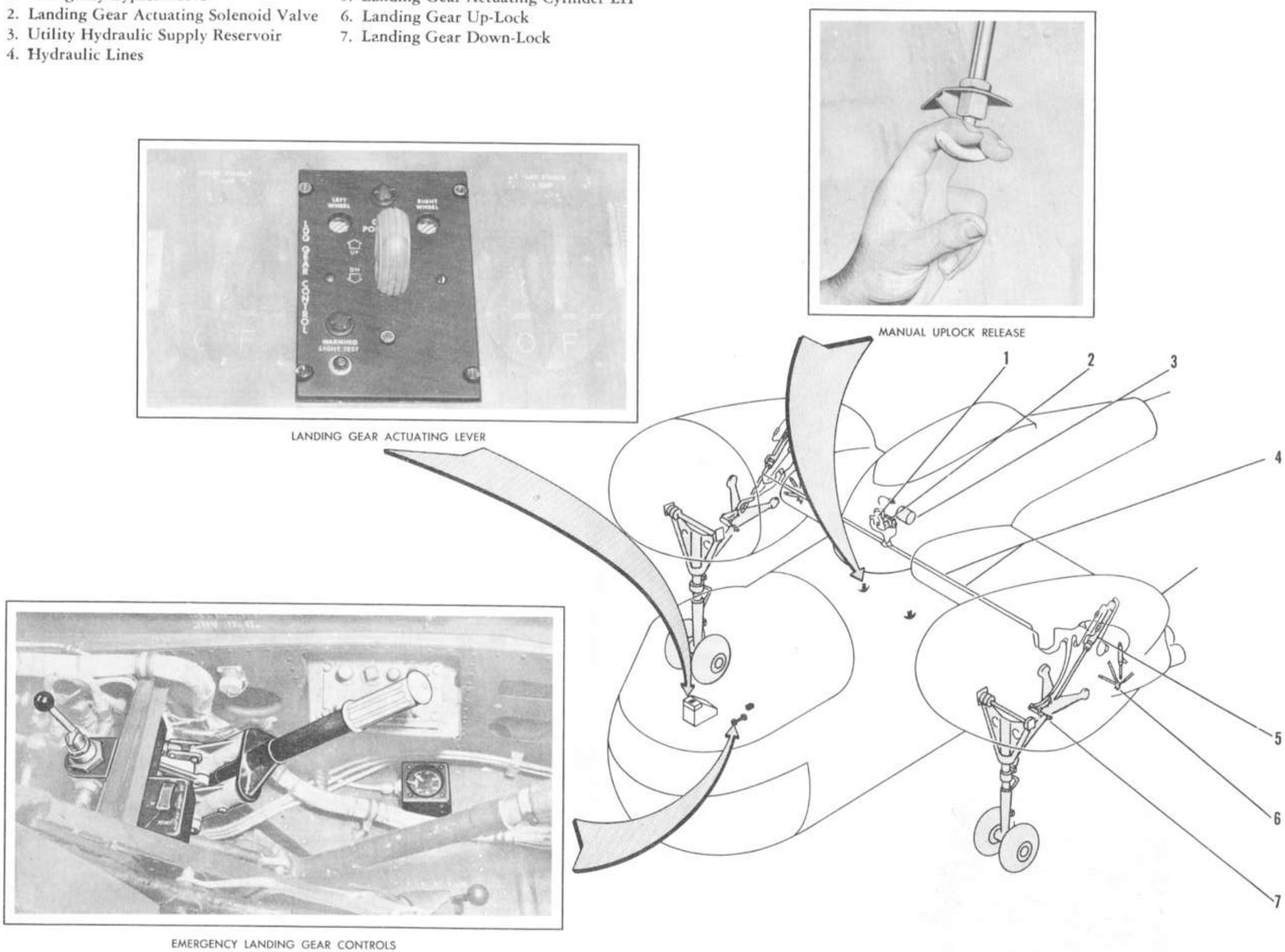


Figure 1—39. Landing Gear System

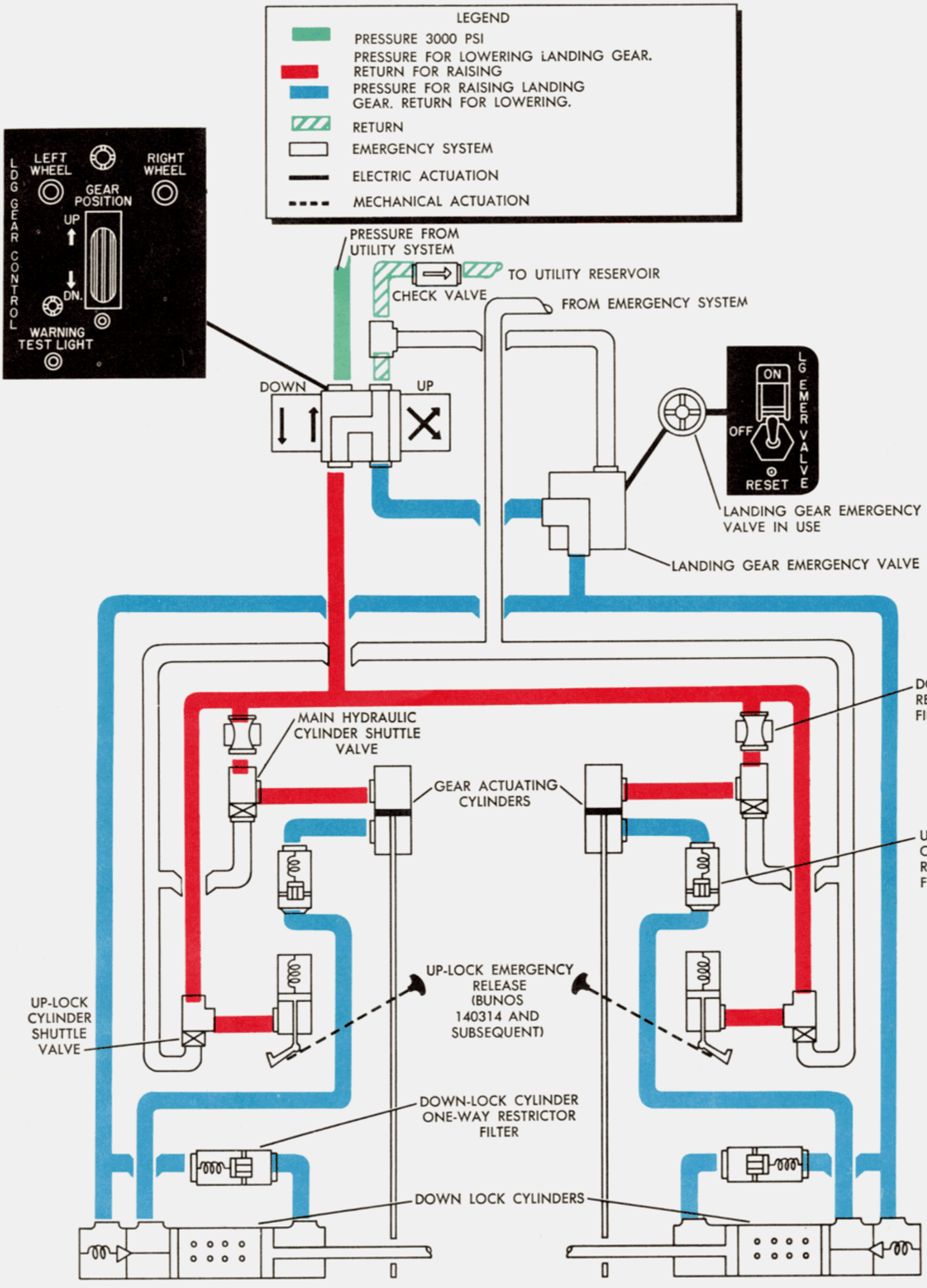


Figure 1—40. Landing Gear System Schematic Diagram

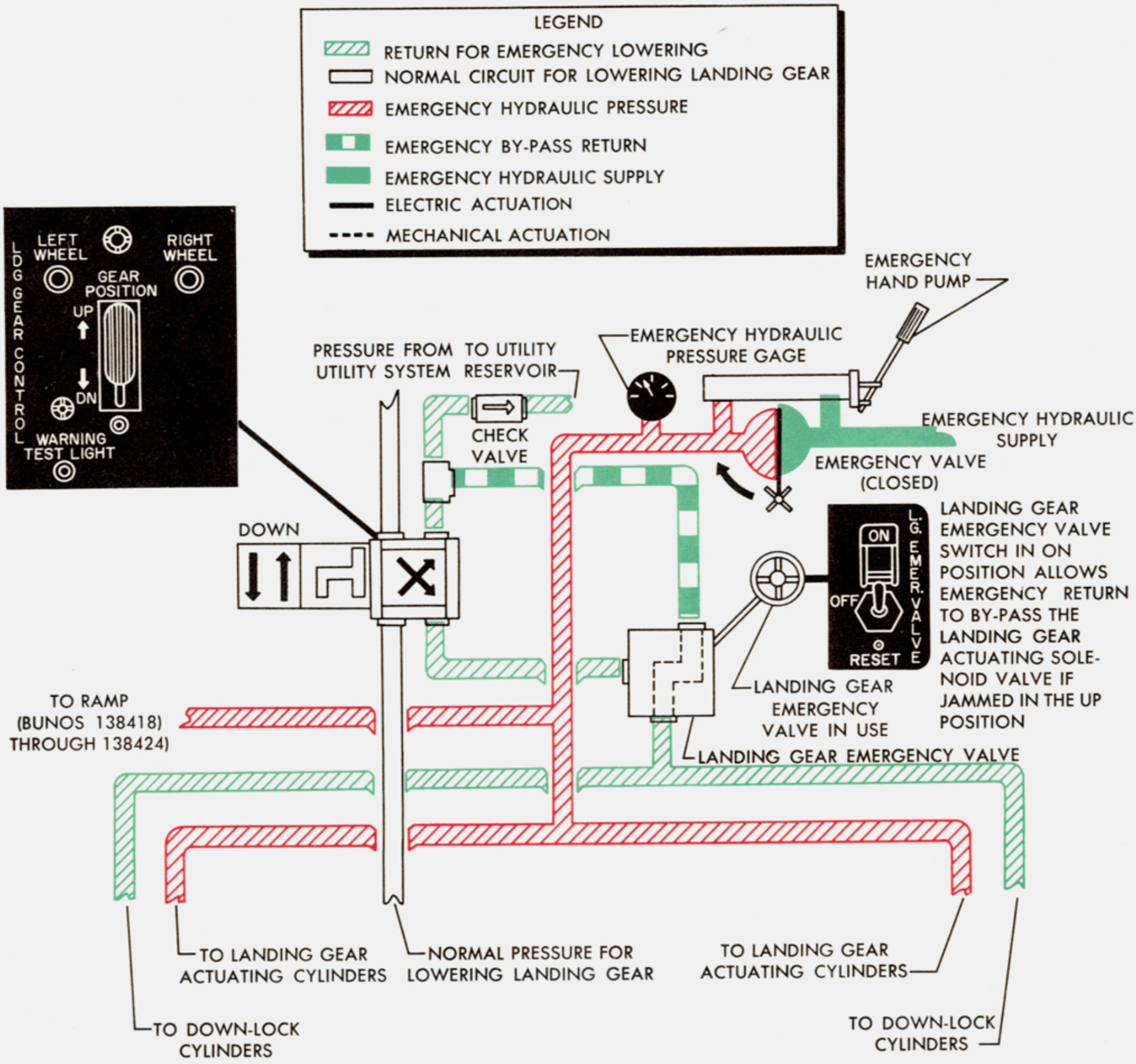


Figure 1—41. Emergency Landing Gear System Schematic Diagram

copter is in the normal ground attitude. The effect of the shortened strut on the take-off attitude of the helicopter is negligible.

LANDING GEAR EMERGENCY VALVE SWITCH.

A guarded landing gear emergency valve switch (8, figure 1—8), marked L.G. EMER. VALVE, with positions ON, OFF, and RESET, is located at the left of the copilot's seat. The switch utilizes direct current from the primary bus to actuate a motor-driven landing gear emergency valve which causes hydraulic oil to bypass the landing gear actuating solenoid valve if it should jam in either the up or the neutral position during flight. The RESET position will return the valve to the normal flight position only when the helicopter is off the ground and the down-lock limit switches are closed.

For further description and operation, refer to figure 1—41 and to **LANDING GEAR FAILURE,** Section III.

LANDING GEAR EMERGENCY VALVE WARNING LIGHT. A red landing gear emergency valve warning light (36, figure 1–12), marked LDG. GEAR EMER VALVE IN USE, located on the bottom left side of the instrument panel, will light when the landing gear emergency valve switch is on.

UP-LOCK EMERGENCY RELEASE.

A right and a left landing gear manually operated, up-lock emergency release permits lowering of the landing gear in case of malfunction of the up-lock mechanism. The two T-shaped release handles (figure 1–39) are located on the upper right and left sides of the cabin walls aft of the rear wing beam. To operate the emergency releases, a crew member pulls the appropriate up-lock release handle.

LANDING GEAR POSITION INDICATORS.

Immediately forward of the landing gear actuating lever are the two main landing gear position indicators (figure 1–39), marked GEAR POSITION, LEFT WHEEL, RIGHT WHEEL. The indicators, which operate on direct current from the primary bus, read UP only if the main landing gear wheels are in the up and locked position, and show pictures of landing gear wheels only if the wheels are in the down and locked position. During landing gear extension and retraction (whenever the landing gear is neither up and locked nor down and locked) and whenever electric power is not available for main landing gear actuation, the indicators show red and yellow diagonal lines.

TAIL WHEEL LOCK LEVER AND INDICATOR.

The tail wheel lock lever (8, figure 1–7) that controls the tail wheel centering lock operates in a quadrant marked TAIL WHEEL, LOCK, UNLOCK, located overhead, to the left of the overhead switch panel. Pushing the lever forward to the UNLOCK position pulls the lockpin from the tail wheel swivel joint and permits the tail wheel to swivel through 360 degrees. Pulling the lever aft to the LOCK position slackens the control cable and permits the spring-loaded lockpin to engage after the tail wheel is centered; as the lockpin engages, a small red flag, marked "Tail Wheel Locked," protrudes from the bottom of the fuselage forward of the tail wheel. The tail wheel lock lever must be pushed to the left to clear detents at each position. After placing the lever in the LOCK position, taxi slowly in a straight line until the tail wheel is centered and locked to avoid shearing the lockpin by rapid swiveling of the tail wheel.

BRAKE SYSTEM.

The main landing gear wheels are equipped with hydraulic boost power brakes connected to the utility hydraulic system and operated by toe pedals on the pilot's tail rotor control pedals. A parking brake is also provided.

BRAKE PEDALS.

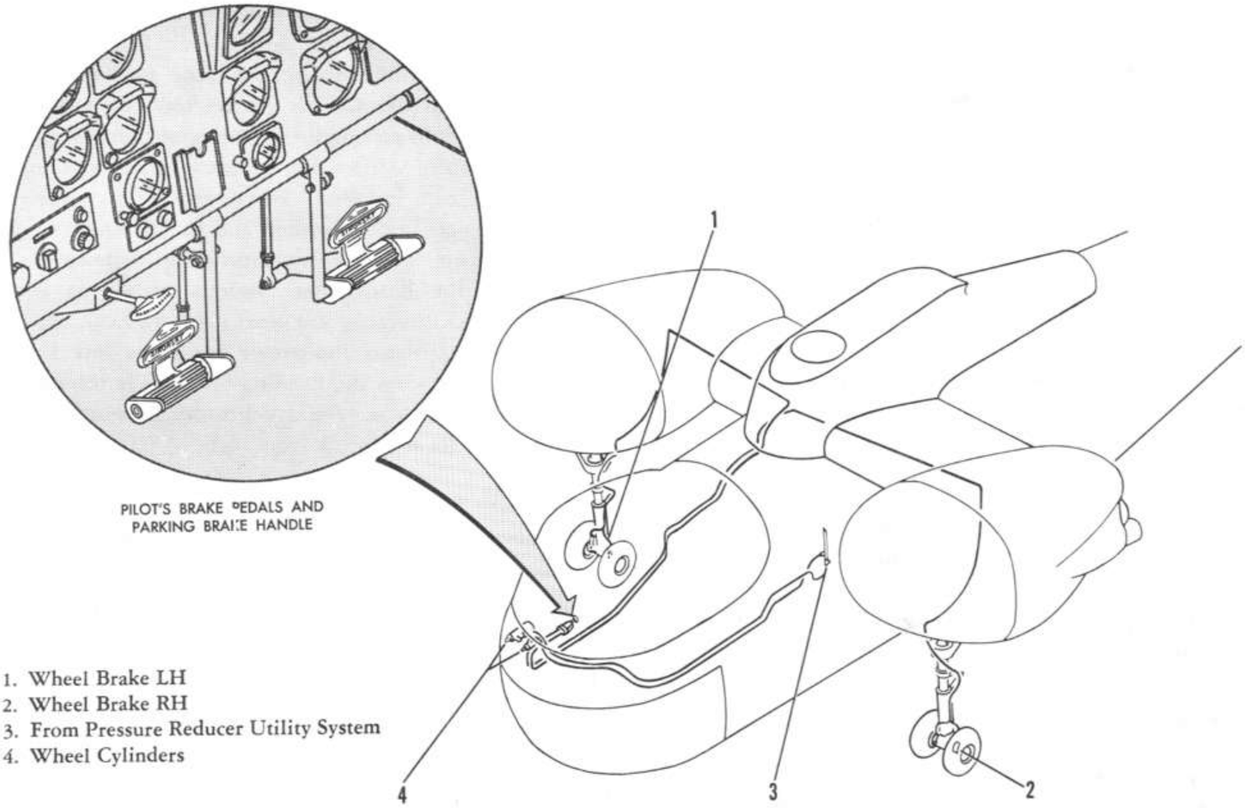
Each main wheel is individually braked by depressing the corresponding wheel brake pedal (6, figure 1–5), mounted above the pilot's tail rotor control pedals. In case of loss of pressure in the utility hydraulic system, the brakes may be applied, but will operate only by pressure developed by the pilot in depressing the brake pedals. The copilot's control pedals are not equipped with brake pedals.

PARKING BRAKE HANDLE.

A parking brake handle (7, figure 1–5), marked PARKING BRAKE, is located below the instrument panel just to the right of the control console. The parking brake is applied by depressing the toe brake pedals, pulling the parking brake handle out, and releasing the pedals while holding the handle out. The handle will remain in the set (on) position. Pressing the toe brake pedals will release the parking brakes, causing the handle to return to the OFF (forward) position.

INSTRUMENTS.

The turn-and-slip indicators (37 and 53, figure 1–12) operate on direct current from the primary bus; all temperature gages operate from the secondary bus. These instruments are protected by circuit breakers located on the overhead circuit breaker panel (figure 1–24). Should both generators fail, as indicated by the lighting of both generator failure warning lights, rendering the secondary bus inoperative, the temperature gages may be restored to operation by placing the battery switch in the BATT EMER position. Turn off all nonessential equipment. The radio magnetic indicators, the attitude indicators, and all pressure gages operate on alternating current from the No. 1 inverter; the radar height indicator operates from the No. 2 inverter. The fuel quantity indicating system and the gyromagnetic compass system operate on direct current from the primary bus and alternating current from the No. 1 inverter; the compass system also obtains power from the No. 2 inverter. Fuses (figure 1–24) for ac instru-



1. Wheel Brake LH
2. Wheel Brake RH
3. From Pressure Reducer Utility System
4. Wheel Cylinders

Figure 1—42. Wheel Brake System

ments are located on a panel on the left side of the control console. Should the No. 1 inverter fail, as indicated by the lighting of the instrument power failure warning light, the ac instruments may be restored to operation by placing the flight instrument power switch in the NO. 2 INV position.

PITOT-STATIC SYSTEM.

Two pitot tubes (20, figure 1—2), one for the pilot's airspeed indicator and one for the copilot's airspeed indicator, are located on the right-hand side of the forward turtle deck. Two static ports (4, figure 1—2) are located in each side of the fuselage aft section. Tubing connects one static port on each side to the pilot's airspeed indicator, vertical velocity indicator, and altimeter. The other pair of static ports is connected to the same instruments in front of the copilot. The pitot tubes may be heated to prevent icing; refer to **PITOT HEATERS,** Section IV.

ATTITUDE INDICATORS.

The MB-1 attitude indicators (3 and 17, figure 1—12) show the flight attitude for any angular displacement of the helicopter in relation to the earth's horizontal plane. A vertical-seeking gyro, linked to a horizon bar, stabilizes a sphere with pitch attitude markings. The sphere and horizon bar are read in combination with an attitude bar against a zero-bank index and a fixed dial with bank markings. The gyros of the indicators are caged by pulling the caging knob at the lower right of the instrument. The caging mechanism allows the gyro to be erected almost instantaneously, whether the gyro motor is spinning or at rest.

WARNING

Since the caging device cages the gyro to the attitude of the helicopter and not to the true vertical, the indicators should not be caged in flight unless the helicopter is known to be in straight and level flight during the caging procedure.

The pilot should not rely upon the attitude indicator if the power warning flag, marked OFF, is visible in the face of the instrument. The attitude indicators operate on alternating current from the No. 1 inverter.

GYRO-MAGNETIC COMPASS SYSTEM.

The GE type MA-1 compass system provides accurate, reliable, and stabilized compass indications by combining the advantages of the remote indicating magnetic compass with the gyro compass. The oscillations of the magnetic compass and the drift error of the directional indicator are eliminated when operating as a gyro-magnetic compass and an accurate stabilized magnetic heading is indicated. In magnetically unreliable regions such as those encountered in high latitudes and on carrier decks, or when making sustained turns, the gyro may be unslaved from the compass system to act as a free directional indicator. The system consists of a remote compass transmitter, located in the aft fuselage section, a directional gyro and amplifier, located in the nose of the helicopter, and a control panel (48, figure 1–12), located on the instrument panel. Directional headings are indicated by the rotating dial face of the radio-magnetic indicators on the instrument panel. The system operates on direct current from the primary bus and alternating current from the No. 1 inverter. Should the No. 1 inverter fail, alternating current is supplied automatically by the No. 2 inverter. The compass system also supplies directional signals to the automatic stabilization equipment.

COMPASS CONTROL PANEL.

The compass control panel (figure 1–44), located on the instrument panel, contains all the controls for the operation of the GE type MA-1 compass system. The selector switch on the compass control panel has the three marked positions: FREE N. LAT, SLAVED, and FREE S. LAT. When the switch is placed in either of the FREE positions, the system will function as a free directional indicator with either north or south latitude corrections for the drift effect of the rotation of the earth. When the switch is placed in the SLAVED position, the directional indicator is slaved to the magnetic compass heading and the rotating dial of the radio-magnetic indicators will indicate stabilized magnetic headings. The heading-set knob, marked PULL TO SET, can be rotated to set up any desired heading on the radio-magnetic indicator and will quickly synchronize the gyro to the magnetic heading of the helicopter when the gyro is slaved. The directional stabilization function of the automatic stabilization equipment is disengaged when this knob is pulled out. The synchronizing indicator is a white pointer visible through a window located directly above the selector switch. When the pointer is in line with the white arrow on the control panel, the system is in synchronization. A red flag on the synchronizing indicator appears whenever electrical power to the system is turned off or has failed. Synchronization is obtained by pulling out the heading-set knob and rotating it until the pointer of the synchronizing indicator is in line with the arrow.

CAUTION

Two settings of the heading-set knob will cause the synchronizing indicator to line up with the arrow. One is correct and the other will result in an unstable 180-degree ambiguity. The correct setting can be recognized by the relationship between the direction that the heading-set knob is turned and the direction of movement of the synchronizing indicator. When the heading-set knob is turned in a clockwise direction, the synchronizing indicator should approach the arrow from the left to right, and when the heading-set knob is turned counterclockwise, the synchronizing indicator approaches the arrow from right to left.

One method of insuring correct synchronization and preventing any possibility of the 180-degree unstable ambiguity is to rotate the heading-set knob in a clockwise direction until the synchronizing indicator moves from left to right, then turn the heading-set knob until the synchronizing indicator lines up with the arrow. The synchronzing indicator will continue to provide a check on the slaving operation during flight; however, the pointer normally will oscillate slightly about the arrow. When the system is being used as a free gyro, the selector switch is set to either FREE N. LAT or FREE S. LAT and the latitude is set on the latitude compensation control, marked SET TO LAT, to compensate for the apparent drift of the gyro due to the rotation of the earth. When the helicopter is being flown in a northerly or southerly direction, the latitude compensation control should be set periodically to the latitude at which the helicopter is flying. The synchronizing indicator has no significance during free gyro operation.

SLAVED GYRO OPERATION.

1. Generator switch (one or both) – ON.
   The power-failure indicator flag should disappear from the synchronizing indicator in about 10 seconds. Allow two minutes more for the equipment to warm up.

2. Selector switch – SLAVED.

3. Heading-set knob – Synchronize gyro and magnetic heading by pulling the knob out and rotating it until the synchronizing indicator is centered.

FREE GYRO OPERATION.

1. Generator switch (one or both) – ON.
   The power-failure indicator flag should disappear from the synchronizing indicator in about 10 sec-

Figure 1—43 deleted.

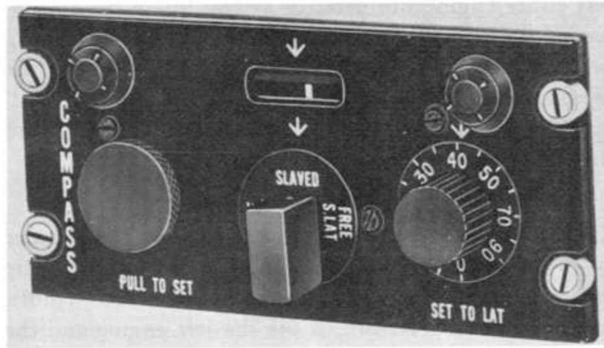


Figure 1—44. Gyro-Magnetic Compass Control Panel

onds. Allow two minutes more for the equipment to warm up.

2. Selector switch — FREE N. LAT or FREE S. LAT — As required.

3. Latitude compensator control — Set to degree of latitude.

4. Heading-set knob — Set to desired heading.

STANDBY COMPASS.

A magnetic standby compass is located at the top of the windshield on the center line of the helicopter. A compass correction card and a radio frequency card are attached to the standby compass mounting bracket. The standby compass light may be turned on or off as desired by a toggle switch at the right of the compass whenever the flight instrument lights are on.

FREE-AIR THERMOMETER.

A bimetallic free-air thermometer (15, figure 1—2) is located in the windshield just below the standby compass. The gage indicates outside air temperature in degrees Centigrade.

CLOCKS.

Two eight-day, 12-hour clocks (39 and 51, figure 1—12) are located on the instrument panel, one in front of the pilot, and one in front of the copilot. Each clock is equipped with a sweep second hand and a totalizer minute hand which may be started, stopped, or reset to zero by successive pushes on a small knob at the upper left of the case. The clocks are wound and set by knobs at the lower right of the cases.

EMERGENCY EQUIPMENT.

ENGINE FIRE DETECTOR SYSTEMS.

Two fire detector systems (one for each engine) are installed to warn the pilot of an engine fire. Three continuous-element fire detector cables are located in each nacelle around the engine accessory shroud, the carburetor air intake, and the oil cooler duct. They are wired into a closed series loop connected to a relay which turns on a warning light in the pilot's compartment in the event of a fire. The fire detector systems operate on direct current from the primary bus.

ENGINE FIRE WARNING LIGHTS AND TEST SWITCHES.

Two red engine fire detector warning lights (50, figure 1—12), one for each engine, are installed on the instrument panel in front of the pilot. The lights are marked FIRE DETECTOR WARN, NO. 1 — ENG. — NO. 2. Two additional red engine fire warning lights, one for each engine, are installed in the engine fire emergency handles, marked FIRE EMERGENCY WARNING, NO. 1 ENGINE, NO. 2 ENGINE, located on the overhead switch panel. A light on the instrument panel and a light in either the left or right engine fire emergency handle will illuminate in the event of a fire in the corresponding engine compartment. Inboard of the emergency handles are two test switches, marked FIRE TEST and ON. To test either engine fire detector system, place the spring-loaded switch in the center position. The corresponding fire warning lights on the instrument panel and in the fire emergency handles should go on. The switch will return to the FIRE TEST position when released and the lights will go out.

ENGINE FIRE EMERGENCY HANDLES.

Two T-shaped engine fire emergency handles (figure 1–10), marked FIRE EMERGENCY WARNING, are located on the overhead switch panel. The handle, marked NO. 1 ENGINE, is for the left engine and the handle, marked NO. 2 ENGINE, is for the right engine. When either handle is pulled down, direct current from the primary bus actuates the fuel shut-off valve and three engine fire emergency valves which close the engine oil and two hydraulic fluid lines leading through the corresponding engine compartment fire wall. This will cause engine stoppage. The ends of the handles house fire detector warning lights. A fire emergency placard, giving procedures to be followed in case of an engine fire, is located on each side of the overhead switch panel.

ENGINE FIRE EXTINGUISHING SYSTEM.

A liquid Bromotrifluoromethane (CF_3Br) one-shot fire extinguisher system is installed to enable the pilot to extinguish an engine fire in either nacelle during flight. The liquid is stored under pressure in a spherical container mounted in the forward left-hand side of the cabin ceiling. Each of two valves on the container contains a disc which is broken by an explosive cartridge when the corresponding fire extinguisher switch is turned on. Tubing extends from one valve to the left engine compartment and from the other valve to the right engine compartment. Within each engine compartment, the tubing divides into perforated distribution lines which extend along both sides of the engine, over the engine accessory section and duct, and over the oil cooler. The extinguishing liquid, when released through the perforations, turns to vapor which smothers the fire. The spherical container is equipped with a pressure gage and a thermal discharge valve which will discharge overboard out the left side of the helicopter, if the temperature of the sphere reaches 96.3° to 104.4°C (208° to 220°F). When the sphere is properly charged, the pressure gage should indicate a value within the range shown on the decal adjacent to the gage.

WARNING

CF_3Br is very volatile but is not easily detected by odor. It is not toxic and can be considered to be about the same as other freons and carbon dioxide, causing danger primarily by reduction of oxygen. The liquid should not be allowed to contact the skin, as it may cause frostbite or low temperature burns because of its very low boiling point.

Note

Although designed primarily for combating an engine fire during flight, the fire extinguishing system may be used on the ground if other fire fighting equipment is ineffectual or not available. Be sure all ground personnel are clear before using the system.

ENGINE FIRE EXTINGUISHER SWITCHES.

Two guarded engine fire extinguisher switches (figure 1–10), marked FIRE EXT. ON, are located on the overhead switch panel in the pilot's compartment. Either switch discharges the contents of the fire extinguisher sphere into the corresponding engine compartment when the guard is raised and the switch placed in the forward position. The switches operate on direct current from the primary bus.

PORTABLE FIRE EXTINGUISHER.

A portable hand-operated CO_2 fog-type fire extinguisher (8, figure 3–4) is installed in the cabin on the right side, aft of the cargo door.

CREW ALARM BELL SWITCH.

A guarded crew alarm bell switch (4, figure 3–4), marked CREW ALARM, ON, and surrounded by an orange stripe, is located on the aft end of the pilot's compartment dome light panel. Lifting the guard and placing the switch in the forward position rings a bell located in the cabin ceiling to alert the occupants of the cabin in case of emergency. The alarm bell is operated from the battery bus and can be sounded even though the electric system is inoperative or shut down.

FIRST AID KITS.

One first aid kit (1, figure 3–4) is mounted on the right side of the cabin aft bulkhead. A second first aid kit (5, figure 3–4) is installed in the pilot's compartment above the emergency hydraulic pump.

PYROTECHNIC PISTOL.

A pyrotechnic pistol (6, figure 3–4) and 12 aircraft signal flares are stowed in a floatable container located at the pilot's left, behind the seat. The container is opened by releasing the latch at the top. The floatable container may be removed from the helicopter by grasping the handle and pulling upward, then inboard. Be sure the cover is latched closed.

LIFE RAFT.

On BuNos 140314 and subsequent, a life raft installation (2, figure 3–4) is provided, to secure a 20-man inflatable life raft to two overhead braces, located in the cabin over the fuselage aft section door. The braces project forward from the cabin bulkhead and support the life raft by means of nylon straps clipped to four ring and eyebolt assemblies. The strap length can be adjusted at the forward ty-clips. The life raft may be released by easing the tension on the sling strap ty-clips supporting the raft against the overhead braces, and releasing the ty-clips from the two forward brace eyebolt rings.

EMERGENCY EXITS.

PILOT'S COMPARTMENT EMERGENCY EXITS.

A panel on each side of the pilot's compartment, composed of the sliding window, the triangular window forward of the sliding window, and the window tracks, can be jettisoned to provide an emergency exit (figure 3–2). The panel will jettison with the sliding window in any position. The energency release handles inside the pilot's compartment, marked EMERGENCY EXIT, PULL, are located outboard of the instrument panel. When the handle is pulled aft, the panel drops outward and downward. The panels can be released from the outside by pulling down on the handle, marked PULL DOWN, EXIT RELEASE, located below the lower sliding window track.

CABIN EMERGENCY HATCH.

An emergency exit (figure 3–2) that contains the aft cabin window on the left side of the fuselage is provided for the occupants of the cabin. The inside emergency release handle, marked EMERGENCY EXIT, PUSH DOWN, is located at the lower aft corner of the hatch and is indicated by a luminescent marker. To open the emergency hatch, swing the handle down and forward and push the hatch out. A handle, marked TURN, EXIT RELEASE, is also provided to open the emergency hatch from outside the helicopter. Turn the handle clockwise and pull the hatch outward.

NOSE DOOR EMERGENCY EXITS.

The upper portion of each nose door can be jettisoned to provide an emergency exit (figure 3–2) from the forward end of the cabin. The release handles inside the cabin, marked EMERGENCY EXIT, PULL TO JETTISON, are located at the top center of each door. When the handle is pulled outboard, the top of the door falls out. The panel may be opened from the outside by pulling out on the handle, marked EXIT RELEASE, PULL TO JETTISON, located above the center window of each door.

PASSENGER DOOR EMERGENCY EXIT.

The upper half of the passenger door, comprising the aft part of the cargo door on the right side of the cabin, may be jettisoned to provide an additional emergency exit (figure 3–2). Pull down on the handle, marked PULL DOWN EMER. EXIT, located at the top of the door, and push outward. The panel may also be opened from outside by turning the latch, marked TURN EXIT RELEASE, at the top and pulling outward.

CABIN WINDOWS.

The eight cabin windows, four on each cabin side panel, may be pushed out to provide additional emergency exits (figure 3–2). EMERGENCY EXIT, PUSH OUT WINDOW is stenciled below each window on the inside. Each of the cabin windows is equipped with a pull tape, outside, at the lower aft corner, marked PULL TAB EXIT RELEASE, by which the locking strip may be pulled out of the rubber seal surrounding the window pane. The panes may then be removed to provide emergency openings. Because of their small size, the cabin windows are not suitable as bailout hatches; use the emergency exits.

PILOT'S COMPARTMENT SLIDING WINDOWS.

A sliding window is located on each side of the pilot's compartment. On BuNos 145856 through 145875, and helicopters modified in accordance with Aircraft Service Change No. 33, a bubble-type window replaces the former flat one. The magnesium tracks are replaced by extruded aluminum tracks and a new lock is provided to permit locking the window in any one of four position in addition to the fully closed position. The window is opened by turning the latch handle counterclockwise and sliding the window aft. The window may be locked by allowing the window lock to engage in the appropriate locking hole in the lower track.

PILOT'S AND COPILOT'S SEATS.

The pilot's and copilot's seats are located side by side in the pilot's compartment with the pilot's compartment access ladder between them. The pilot's seat is on the right. The seats are designed to accommodate back-type parachutes and pararafts. Both seats have a 5-inch range of height adjustment and are equipped with safety belts and shoulder harnesses. The seats are also equipped with seat and back cushions.

SEAT HEIGHT ADJUSTMENT LEVERS.

A seat height adjustment lever below the right side of each seat is pulled up to release the seat lockpins. The seat aided by a spring-loaded bungee can then be adjusted for height by varying the weight upon it. The lockpins will automatically engage in any of six positions when the adjustment lever is released.

SHOULDER HARNESS INERTIA REEL LOCK LEVER.

A two-position shoulder harness inertia reel lever (6, figure 1–8) is located below the left side of each seat.

1. Auxiliary Fuel Tanks (Left and Right)
2. Fuel Tanks (Left and Right)
3. Second Stage Servo Hydraulic Reservoir
4. Utility Hydraulic Reservoir
5. Main Gear Box
6. First Stage Servo Hydraulic Reservoir
7. Rotor Brake Accumulator
8. Intermediate Gear Box
9. Tail Gear Box
10. Engine Oil Tanks (Left and Right)
11. Rotor Clutch Oil Reservoir (Left and Right)
12. Engine Fire Extinguisher Container

Fuel	MIL-F-5572, Grade 115/145.
Engine Oil	MIL-L-6082, Grade 1100(s). Grade 1065(w). Use grade 1100 oil for continuous temperatures above 25°F (—4°C) and grade 1065 oil for continuous temperatures below 25°F (—4°C). Preheat, if grade 1065 oil is not available, and oil dilute if preheat is not available.
Main Gear Box Oil	MIL-L-9000 MS-9250 (s and w) or alternate MIL-L-21260 Grade 2 (s and w) down to 0°F (—18°C). Below 0°F (—18°C) use auxiliary heat to raise to 0°F (—18°C).
Intermediate Gear Box Oil	MIL-L-7808, (s and w).
Tail Gear Box Oil	MIL-L-9000 MS-9250 (s and w) or alternate MIL-L-21260 Grade 2 (s and w) down to 0°F (—18°C). Below 0°F (—18°C) use MIL-L-7808.
APU Oil	MIL-L-6082, Grade 1100(s). Grade 1065(w). Use grade 1100 oil for continuous temperatures above 25°F (—4°C) and grade 1065 oil for continuous temperatures below 25°F (—4°C). Preheat, if grade 1065 oil is not available, and oil dilute if preheat is not available.
Rotor Clutch Oil	MIL-L-21260, Grade 2, down to 32°F (0°C). Below 32°F (0°C) use MIL-O-6081, Grade 1010.
Hydraulic Fluid	MIL-H-5606.
Engine Fire Extinguisher Agent	Bromotrifluoromethane (CF3Br).

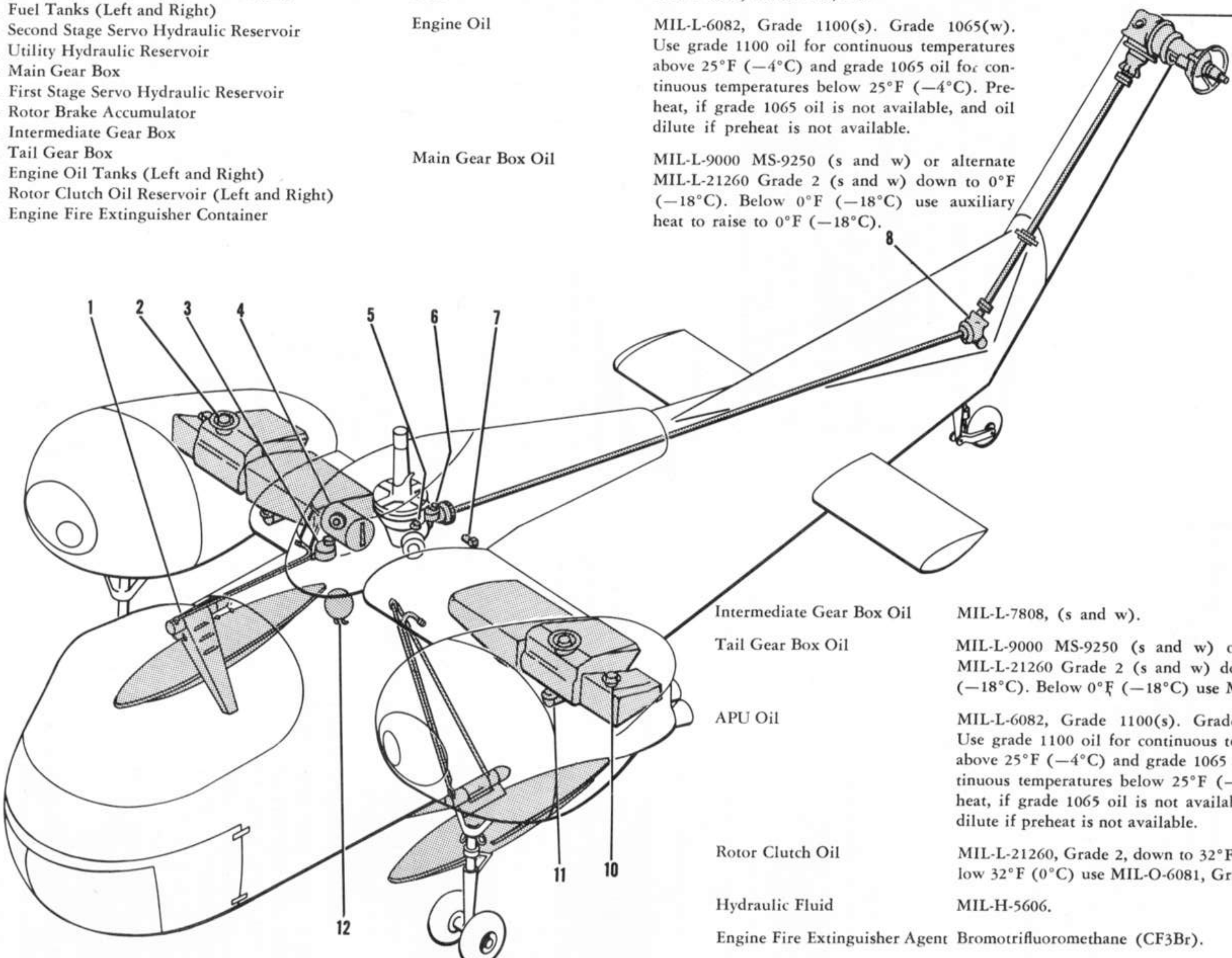


Figure No. 1—45. Servicing Diagram (Sheet 1)

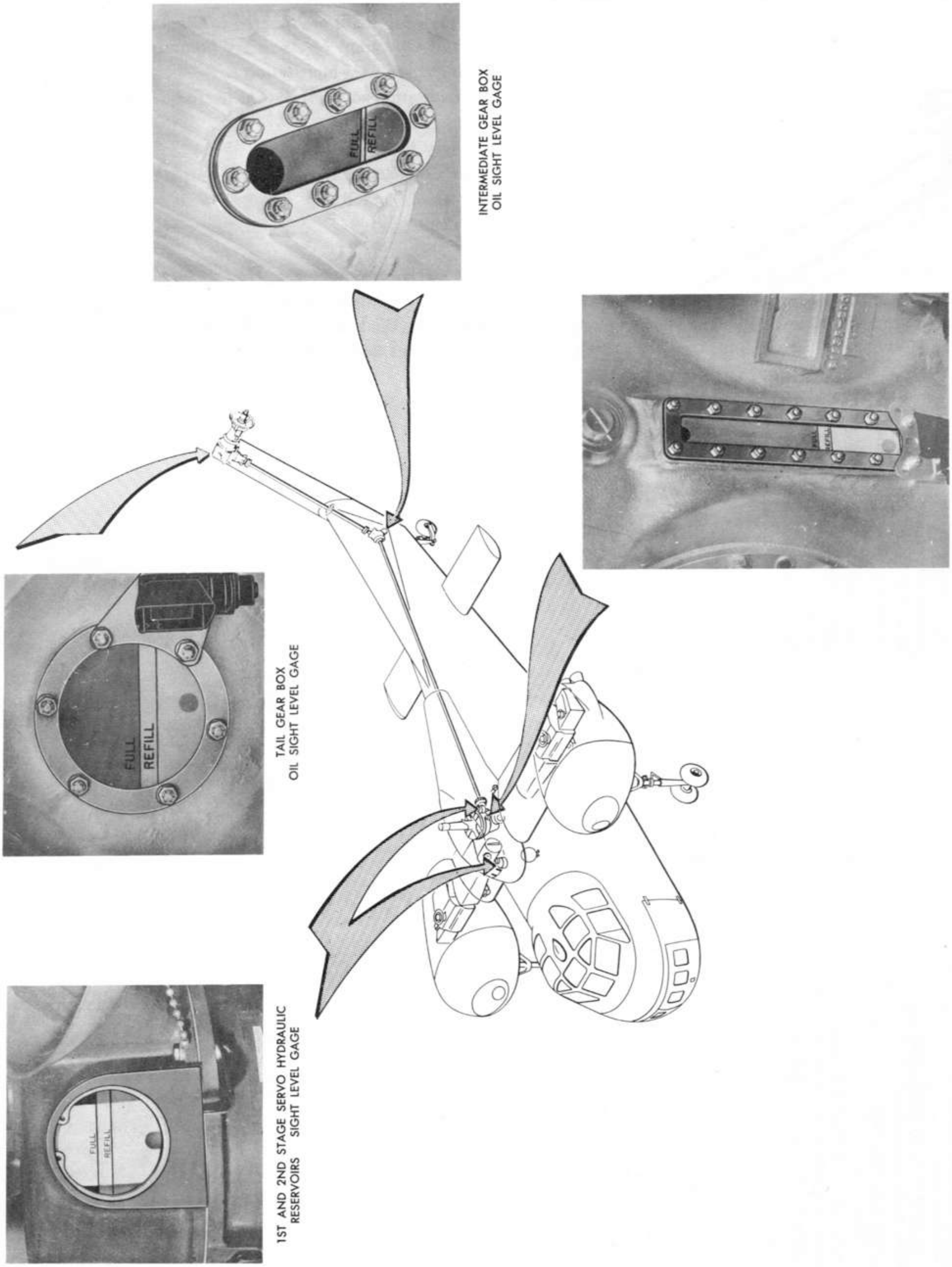


Figure No. 1—45. Servicing Diagram (Sheet 2)

By pressing down on the top of the lever, a latch is released and the lever may be moved from one position to the other. When the lever is in the UNLOCKED (aft) position, the shoulder harness cable will extend to allow the occupant to lean forward, however, the cable reel will automatically lock if an impact force of two or three g's in a forward direction is encountered. The cable reel will remain locked until the lever is moved to the locked position and then returned to the UNLOCKED position. When the lever is in the LOCKED (forward) position, the shoulder harness cable is locked so that the occupant is prevented from leaning forward. The LOCKED position is used when a crash landing or ditching is anticipated to provide an added safety precaution over that of the automatic lock, though it may be used at take-off or landing.

AUXILIARY EQUIPMENT.

The following items are covered in Section IV, Description and Operation of Auxiliary Equipment:

Heating System (Winterization Kit)
Communication and Associated Electronic Equipment
Lighting Equipment
Auxiliary Power Unit
Cargo Compartment
Nose Doors and Ramp
Cargo Hoist
Cargo Sling
Troop Carrying Equipment
Casualty Carrying Equipment
Main Rotor Blade and Pylon Folding
Miscellaneous Equipment

SECTION II
NORMAL PROCEDURES

BEFORE ENTERING THE HELICOPTER.

FLIGHT RESTRICTIONS.

(Refer to Section V.)

CRUISE CONTROL.

Required fuel, power settings, airspeed, take-off, climb, cruising flight, and landing data may be determined by use of the operating data charts in Appendix I.

WEIGHT AND BALANCE.

It is possible to load this helicopter outside safe cg limits. Before take off, check take-off, in-flight, and landing gross weights and cg locations by referring to the Handbook of Weight and Balance, AN 01-1B-40, for loading information. For cg limits, refer to **CENTER OF GRAVITY LIMITATIONS,** Section V. A balance computer is stowed at the right of the copilot's seat.

EXTERIOR INSPECTION.

Consult yellow sheets for the status of the helicopter. Perform exterior inspection of the helicopter in accordance with figure 2–1.

ENTRANCES TO THE HELICOPTER.

The cabin is entered through a two-piece door (26, figure 1–2) which forms the aft part of the cargo door on the right side of the fuselage. The lower half of the door forms a step when hinged down and the upper half swings upward and is supported by struts. The cabin may also be entered through the hydraulically operated nose doors (17, figure 1–2) if they are open; however, they are used primarily for troop or cargo loading.

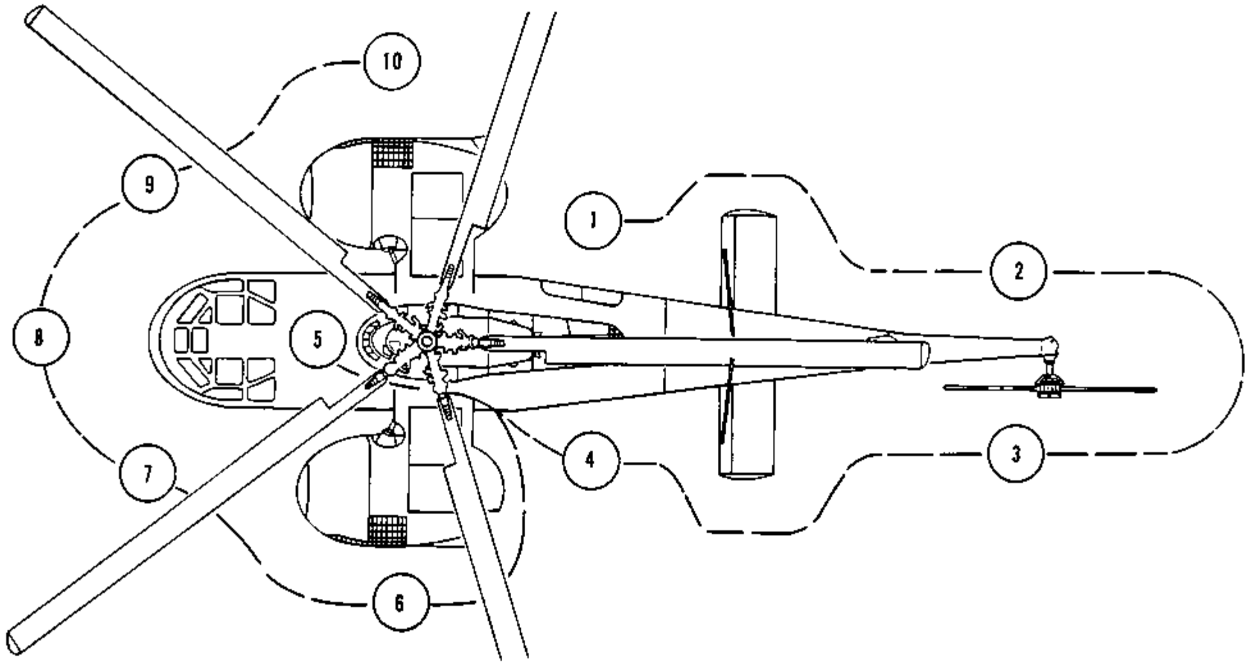
The pilot's compartment may be entered by means of a ladder (8, figure 1–3) at the forward end of the cabin. The ladder is hinged at its upper end. The pilot's compartment may also be entered from outside the helicopter by climbing up the pilot's compartment access footwells (19, figure 1–2) aft of the nose door, and entering through the sliding window on the right side of the helicopter.

ON ENTERING THE HELICOPTER.

INTERIOR CHECK (ALL FLIGHTS).

CABIN.

1. Cabin cargo and passenger doors – Closed.
2. First aid kit – Complete and secure.
3. APU – Secure, no fuel leakage.
4. Portable fire extinguisher – Charged and secure.
5. Stabilizer manual override valve – Secure.
6. Stabilizer shaft and bracket bearings – Secure.
7. Control cables and leads to AN/ARC-2A (flashlight required) – Clear.
8. Aft cabin bulkhead door – Closed.
9. Battery bus and high amperage circuit breakers (overhead) – Set.
10. Troop seats, litters, or cargo – Secure.
11. Cargo hoist winch – Locked to rail forward of curved track, within the decaled limits.
12. Cabin emergency hatch – Secure and accessible.
13. Canteens – Filled and secure.
14. Engine fire extinguisher sphere (overhead) – Properly charged and secure.
15. Tail rotor drive shaft – Secure.
16. Flight control rods – Clear.
17. Cargo sling stowage line – Sling stowed, line secured.
18. Gear box inspection light switch – OFF.
19. Auxiliary fuel tank jettison handles and covers – Secure.



1. RIGHT REAR OF FUSELAGE.

a. Rotor blades for tears or loose pockets.
b. Main gear box cooling air and heater vent and exhaust openings for freedom from obstruction.
c. Fuselage for dents or tears in skin.
d. Static ports for freedom from obstruction.
e. Cargo door for security and damage.
f. Stabilizer for security and damage.
g. Stabilizer damper strut for security and signs of leakage.
h. Condition of antennas.
i. Intermediate gear box oil level.
j. Pylon hinges for security and damage.
k. Cotter pins in tail rotor hydraulic swivel joints.

2. RIGHT SIDE OF PYLON.

a. Tail wheel tire for cuts, bald spots, slippage, and proper inflation.
b. Tail wheel oleo strut for damage, leaks, and proper extension.
c. Static ground wire for good contact with ground.
d. Pylon and fin for dents or tears in skin.
e. Pylon fold jury strut secure.
f. Condition of whip antenna.
g. Tail rotor gear box oil level.

3. LEFT SIDE OF PYLON.

a. Tail rotor head for security of all components.
b. Tail rotor blades for dents or tears.
c. Tail rotor blades coned.
d. Pylon and fin for dents or tears in skin.
e. Intermediate gear box cooling air openings for freedom from obstruction.
f. Signs of leakage from intermediate or tail gear boxes.
g. Pylon lockpin indicator retracted.
h. Pylon lockpin ratchet handle secure.
i. Pylon fold switch in center position and access door secure.

4. LEFT REAR OF FUSELAGE.

a. Make the same inspection as on the right side, except for steps b., e., and h. through j.
b. APU exhaust openings for freedom from obstruction.
c. Fuel leaks at heater and APU vent lines.
d. Cabin emergency hatch secure.

5. AT MAIN ROTOR HEAD.

a. Rotor head for security of all components. If blades have been folded and unfolded, check that bicycle clips have been removed from the dampers. Refer to MAIN ROTOR BLADE AND PYLON FOLDING, Section IV.
b. Droop stops in position.
c. Blade dampers for security and leakage. Check damper oil levels.
d. Signs of leakage from main gear box or main rotor servo units. Check gear box and first stage servo reservoir oil levels.
e. Servo unit boots and control rods for internal ice. Boots for deterioration.

WARNING

Do not operate the helicopter with damaged or deteriorated servo boots because of the danger of water being trapped around the servo pilot valves, causing corrosion or freezing of the flight controls.

f. Second stage servo and utility reservoir oil levels.
g. Condition of loop antenna.
h. Main gear box fairing and ports for security, damage, and freedom from obstruction.
i. Fuel levels and security of fuel filler caps.

Figure No. 2–1. Exterior Inspection (Sheet 1)

j. Oil levels and security filler caps.
k. Upper surface of blades for tears or loose pockets.
l. Main gear box service platform secure.

6. LEFT WING AND NACELLE.

a. Under surface of wing for security and damage.
b. Nacelle fairings for security and damage.
c. Oil dilution valve in closed position.
d. Engine access door for security.
e. Carburetor cold air intake for freedom from obstructions.
f. Engine cooling air intakes and outlet openings for security, damage, and freedom from obstructions.
g. Landing gear components for security, damage, and leaks. Check down-lock pin visible.
h. Tires for cuts, bald spots, slippage, and proper inflation.
i. Chocks in place.
j. Auxiliary fuel tanks, support structure, cables, and hoses for security, damage, or leakage.
k. Signs of oil, fuel, or hydraulic fluid leaks on understructure or on ground.
l. All drain ports for freedom from obstruction.

7. LEFT FRONT OF FUSELAGE.

a. Pilot's compartment sliding windows and tracks for security of attachment.
b. Fuselage for dents or tears in skin.
c. Rotor blades for tears or loose pockets.
d. Fire extinguisher thermal discharge disc unbroken.

8. NOSE.

a. All canvas protective covers removed.
b. Ramp up and locked, nose doors closed and locked.
c. Landing light for security and damage.
d. Windshield clean.

9. RIGHT FRONT OF FUSELAGE.

a. Make same inspection as on the left side except step d.
b. Pitot tube cover removed; pitot tube free from obstruction.

10. RIGHT WING AND NACELLE.

a. Make same inspection as on the left side.
b. Cargo sling secure, undamaged, and stowed.

Figure No. 2–1. Exterior Inspection (Sheet 2)

20. Throttle servo — Check for leaks, cover, and binding.
21. Blade fold, nose door, and ramp manual override controls — Safety collars in place.
22. Ramp up — Nose doors closed and manual catch secure.
23. Ramp extensions — Secured to cabin sides.
24. AC transformer fuses (forward, overhead) — Secure.

PILOT'S COMPARTMENT.

1. Radio circuit breakers (aft face of console) — Set.
2. Pyrotechnic pistol — Properly stowed; container latched.
3. Circuit breakers (overhead panel) — Set (check while climbing ladder).
4. First aid kit (copilot's left) — Complete and secure.
5. Pilot's seat adjustment lever — Adjust seat height, check security.
6. Tail rotor control pedal adjustment knob — Adjust tail rotor pedals.
7. Safety belt and shoulder harness — Adjust and lock.
8. Shoulder harness inertia reel lock lever — Check operation of inertia reel in both the LOCKED and UNLOCKED positions.
9. Parking brake handle — LOCKED.
10. Dome light rheostat — OFF.
11. Dome light switches — OFF.
12. Crew alarm bell switch — Check operation of alarm bell.
13. Ramp switch — OFF.
14. Nose door switch — OFF.
15. Cargo sling master switch — SAFE. Switch guard closed.
16. Pitot heater switch — PITOT HEAT (off).
17. Auxiliary fuel tank jettison switches — OFF. Switch guards closed.
18. Auxiliary fuel pump switches — OFF.
19. Radio master switch — OFF.
20. Stick trim master or beeper trim master switch — ON.
21. Exterior light switches — OFF.
22. Fuel booster pump switches — OFF.
23. Oil dilution switches — OFF.
24. Fuel cross-feed switch — CLOSE.
25. Flight instrument light rheostat — OFF.
26. Nonflight instrument light rheostat — OFF.
27. Console and panel light rheostat — OFF.
28. Engine fire extinguisher switches — OFF. Switch guards closed, engine fire emergency handle full up.
29. Ignition switches — OFF.
30. Rotor clutch pump switches — OFF.
31. Generator switches — OFF.
32. Battery switch — ON.
33. APU — START.

For operation of APU, refer to **AUXILIARY POWER UNIT,** Section IV.

34. Warning lights – CHECK ALL PRESS-TO-TEST. Generator warning lights will be on until generators reach operating speed and generator switches placed in ON position.
35. Fire test switches – Hold in the ON position. Fire emergency warning lights should go on after a few seconds.
36. Tail wheel lock lever – LOCKED.
37. Rotor brake lever – ON.
38. Mixture control levers – IDLE CUT-OFF.
39. Twist-grip throttle – Rotate through full travel.
40. Control quadrant throttles – CLOSED and LOCKED.
41. Carburetor air levers – COLD.
42. Landing gear emergency valve switch – OFF.
43. Emergency hydraulic shut-off valve – OPEN.
44. Nose door lock lever – LOCKED.
45. Radio switches – OFF.
46. Main rotor blade folding master switch – OFF. Switch guards closed.
47. Safety valve switch – CLOSED.
48. Blade fold-unfold switch – OFF.
49. Stabilizer positioning switch – HOVER. Hovering warning light should be on.
50. Fuel shut-off switch – OFF.
51. Landing gear actuating lever – DN.
52. Landing gear position indicators – Wheels down.
53. Collective pitch control – Check minimum.
54. Landing light switches – OFF.
55. Flight instrument power switch – NO. 1 INV.
56. First and second stage servo shut-off switch – FIRST STAGE OFF.
57. Automatic stabilization servo shut-off switch – ON.
58. Transmission oil low pressure warning light (main gear box) – Should be on.
59. Fuel quantity gage test switches – Check fuel quantity gages for proper operation.

INTERIOR CHECK (NIGHT FLIGHTS).

1. Anti-glare curtains – Installed.
2. Flashlights – Available, red lenses installed.
3. All interior lights – Check operation.
4. All exterior lights – Check operation.

BEFORE STARTING ENGINES.

FUEL SYSTEM PRESSURE CHECK.

1. Fuel shut-off switches – OFF.
2. Fuel booster pump switches – NORM.
3. Fuel pressure gages – Check for zero pressure. Any increase in fuel pressure indicates a leaking fuel shut-off valve.
4. Fuel shut-off switches – ON.
5. Fuel pressure gages – Check for rise of 11 to 16 psi.
6. Engine fire emergency handles – PULL OUT.
7. Fuel booster pump switches – OFF.
8. Fuel pressure gages – Check for steady pressure or gradual decrease.
9. Engine fire emergency handles – PUSH IN.
10. Fuel pressure gages – Check for rise of 17 to 25 psi.
11. Fuel booster pump switches – OFF. Check fuel pressure gages. A drop in fuel pressure when the pumps are turned off indicates a leaking check valve or mixture control. If fuel pressure drops rapidly, move the fuel shut-off switch to OFF. If pressure drop ceases, it indicates a leaking check valve and that the leakage has been stopped by the shut-off valve. Continued loss of pressure indicates a leaking mixture control. A slow drop to 0 psi is normal, due to bleed holes in the check valve.
12. If the booster pump pressures hold after turning off the booster pumps, move mixture controls to NORMAL. Fuel pressure should drop to the fuel feed valve setting of 8 to 10 psi. A drop in pressure to near zero indicates a sticking fuel feed valve.

Note

To conduct this step, fuel pressure must be above the fuel feed valve setting of 8 to 10 psi.

FUEL CROSS-FEED SYSTEM CHECK.

1. Mixture control levers – IDLE CUT-OFF.
2. Fuel shut-off switches – ON.
3. Left fuel booster pump switch – NORM.
4. Right fuel booster pump switch – OFF.
5. Fuel pressure gages – Check for a pressure differential.
6. Fuel cross-feed switch – OPEN.
7. Fuel pressure gage – Check for approximately same pressure.
8. Right fuel booster pump switch – EMER.
9. Fuel pressure gages – Check for an increase to approximately the same pressure.
10. Fuel booster pump switches – OFF.
11. Fuel cross-feed switch – CLOSE.
12. Fuel shut-off switches – OFF.

STARTING ENGINES.

WARNING

Collective stick friction should be on and the collective stick held down when starting engines as the ASE might inadvertently be hard over. A hard over from ASE in the up collective direction could cause the stick to rise, thereby overspeeding the engines.

Start the left engine first. The utility hydraulic pump is driven by the left engine which supplies hydraulic pressure for the operation of the nose doors and ramp and the hydraulic phase of the main rotor blade and pylon unfolding operation.

Note

For the procedure to be followed in the event of fire while starting engines, refer to **FIRE,** Section III.

1. Fire guard – POSTED.
2. Collective pitch control – MINIMUM PITCH.
3. Twist-grip throttle – Check 40 degrees on twist-grip throttle indicator.
4. Left control quadrant throttle – FULLY CLOSED and LOCKED.

Note

The microswitches on the control quadrant throttles render the starters inoperative unless the throttles are locked in the FULLY CLOSED and LOCKED position. The locked position permits throttle movement up to approximately 1600 rpm, which is sufficient for warm-up procedures.

5. Left fuel shut-off switch – ON.
6. Left fuel booster pump switch – NORM.
7. Left starter button – DEPRESS.

CAUTION

A ground crew member (fire guard) should stand by the engine to signal the pilot if any signs of starter hesitation or overspeeding occur. Should signs of abnormal operation occur, disengage starter, and place fuel shut-off switch and fuel booster pump in the OFF position. Proceed as directed in HYDRAULIC LOCK, Section VII.

CAUTION

If the pistons squeak when the engine is being turned over, it is an indication that the oil film has drained from the cylinder walls. Dry cylinders may cause scoring of the cylinder walls and piston seizure. When this squeaking is heard, do not attempt to start the engine until the spark plugs have been removed and the cylinder walls sprayed with fresh oil.

8. Left ignition switch – BOTH.
   After at least 10 seconds starter operation for cold engine; 5 seconds if the engine has been operated within the last hour.

9. Left primer switch – DEPRESS.
Prime, using short pulses if engine is warm; longer sustained pulses for cold engine. Overpriming can be recognized by excessive drainage from intake pipe drain lines. (See UNDERPRIMING and OVERPRIMING in Section VII.)

CAUTION

Continuous direct cranking of the starter should not exceed 1 minute. If the engine does not start after 30 seconds, allow at least 1 minute for the starter to cool.

CAUTION

Do not pump the throttle. It frequently causes backfiring which may result in damage to the induction system, or an engine fire. Pumping the throttle will also disengage the starter circuit.

10. Left mixture control lever – RICH.
Prime as required until engine operates smoothly at 1000 rpm; then move mixture control lever slowly to RICH. Discontinue priming when engine rpm begins to drop.

CAUTION

- If priming is discontinued before the carburetor is metering normally, excessive leanness may cause backfiring and serious damage to the engine. Backfiring as well as quick accelerations and decelerations should be avoided at all times with all engines. Their effect on this engine is particularly serious with the rotor clutch disengaged and without the benefit of the loading usually provided by a rotor or propeller.
- If for any reason the engine should stop, immediately move the mixture control to IDLE CUT-OFF and continue to operate the starter and primer until the engine starts.

WARNING

Do not open the control quadrant throttle to clear the engine while the ignition switch is on, as serious overspeeding and destruction of the engine could result if the engine should fire and start to run.

11. Engine oil pressure gage – CHECK.
After the engine starts, check oil pressure gage for a rise in oil pressure.

CAUTION

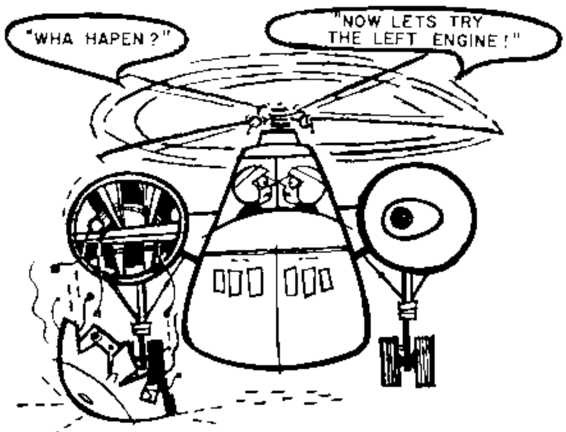
If oil pressure does not register on gage almost immediately, stop the engine and investigate.

Note

If a start is not made within a reasonable time, an investigation should be made to ascertain the cause.

WARNING

Without rotor loading, engine speed is very sensitive to throttle movement. Care must be taken to prevent overspeeding of the engine.



IT'S VERY EASY TO OVER-REV DURING GROUND OPERATION WITHOUT THE ROTOR CLUTCH ENGAGED. OPEN THE QUADRANT THROTTLES SLOWLY AND A VERY LITTLE AT A TIME!

12. Fuel booster pump switch – OFF.
13. Engine-driven fuel pump – CHECK.
14. Left control quadrant throttle – 1000 ENGINE RPM.
15. Generator switch – ON.
If the voltage needle of the volt-ammeter does not move, increase rpm slowly until approximately 28 volts are indicated. Do not exceed 100 psi oil pressure.
16. Utility hydraulic pressure gage – CHECK FOR NORMAL PRESSURE.
17. Follow the same procedure for starting the right engine, except for checking the utility hydraulic pressure.

CAUTION

If an external power unit is used for starting, do not place it under the right engine fuel drain. This is to avoid a fire hazard when starting the right engine as fuel may drain onto the external power unit.

18. After starting right engine, check second stage servo hydraulic pressure gage for normal pressure.

STARTING PROCEDURE USING BATTERY.

If neither external power nor the APU is available, start the left engine, using the normal starting procedure. Power will be supplied by the battery. After starting the left engine, the right engine can be started on power developed by the left generator by proceeding as follows:

1. Left control quadrant throttle – APPROXIMATELY 1300 ENGINE RPM. Advance until the left engine volt-ammeter indicates an output of approximately 28 volts. Do not exceed 100 psi oil pressure. Sufficient direct current will then be supplied to start the engine.
2. Start the right engine using the normal starting procedure while maintaining a 28-volt output of the left generator.

Note

After the left engine is started, auxiliary servo pressure will be available and may cause the tail rotor control pedals to creep. This will happen only if, at shutdown after the previous flight, the required electric power was removed before the automatic stabilization equipment was placed in standby. The pedal creeping can be overcome with moderate pressure on the pedals and will disappear as soon as full automatic stabilization power is available and the equipment is initially engaged.

SECOND STAGE SERVO HYDRAULIC SYSTEM CHECK.

1. Second stage servo hydraulic pressure gage – Check for pressure within the normal operating range at 1000 engine rpm.
2. Servo shut-off switch – SECOND STAGE OFF.
3. Second stage servo hydraulic pressure gage – Check for no decrease in pressure. The pressure switch in the inoperative first stage servo system should prevent the second stage system from being shut off.
4. Servo shut-off switch – FIRST STAGE OFF.

ENGINE GROUND OPERATION.

CAUTION

Prolonged ground operation with the clutches disengaged is not recommended; however, when necessary, engine speed should be held to between 1500 and 1600 rpm to minimize the possibility of the clutches overheating. For additional limitations on ground operation due to clutch overheating, refer to GROUND OPERATION – CLUTCHES DISENGAGED, Section V.

WARM-UP.

1. Control quadrant throttles – 1500 RPM. Do not exceed 100 psi oil pressure.
2. Mixture control levers – RICH.
3. Carburetor air levers – CHECK PROPER OPERATION. Move levers to HOT and check for an increase in carburetor air temperatures; move levers to COLD and check for a decrease in carburetor air temperatures.
4. Auxiliary fuel pump switches – CHECK PROPER OPERATION.
5. Continue engine warm-up at 1500 rpm until engine oil temperature reaches a minimum of 40°C.
6. Flight controls – CHECK. Movement free and correct.
7. Cyclic control stick trim system – CHECK.
   a. Cyclic control stick – CENTERED.
   b. Cyclic control stick master switch – ON. Check for resistance to movement of cyclic control stick in all directions. Resistance should increase as displacement of the stick is increased.
   c. Cyclic control stick – CENTERED. Depress cyclic control stick trim release switch. No resistance should be felt to movement of the cyclic control stick in any direction.
   d. Cyclic control stick trim release switch – DEPRESS. Move cyclic control stick to full left position. Release trim release switch. Stick should remain in position.
   e. Cyclic control stick – MOVE TO FULL OPPOSITE POSITION. Check that resistance to movement increases as displacement increases. Return stick to starting (full left) position.

CAUTION

It is permissible to move the cyclic control stick from one extreme position to another extreme position with the stick trim system operating, but the trim release switch must not be pressed under those conditions. Trimming the cyclic control stick in one extreme position and then moving the stick to the opposite extreme position causes maximum extension of the force gradient spring. When the cyclic control stick trim release switch is depressed, spring tension is released. The force created by releasing the spring at maximum extension may cause damage to the flight control system.

f. Perform the check outlined in steps d. and e. starting with the cyclic control stick in each of the following positions: full right, full forward, full aft.

g. Cyclic control stick trim release switch – DEPRESS.
Move stick to the center position. Release switch.

8. Idle speed check – Engines should idle at 1000 rpm when normal operating temperatures are reached.
9. Ignition switch check – Adjust throttles to 1000 rpm (accomplish this check on left engine first, then on right engine). Switch ignition from BOTH to R and back to BOTH. Switch ignition from BOTH to L and back to BOTH. Switch ignition to OFF (momentarily) and back to BOTH. A slight drop in rpm when operating on L and R magneto and complete cutting out of the engine at OFF position indicates proper connection of the ignition ground leads.

CAUTION

- Perform this check as rapidly as possible to prevent severe backfiring when the switch is returned to BOTH.
- Ground operation should be kept to a minimum with the rotor clutches disengaged as the tendency toward spark plug fouling increases greatly during engine operation with relatively light loading. If such an operation is unavoidable, manually lean the mixture to obtain 25 rpm drop from best power.

OPERATION AND CHECKS OF AUXILIARY GROUND EQUIPMENT.

The main rotor blade folding system and the ramp and nose door actuating system may be operated at 1200 to 1500 engine rpm; however, less pressure is available at low engine rpm and the operations take a longer period of time. The utility hydraulic pump, which supplies pressure for the utility hydraulic system, is driven by the left engine. If the blades and pylon are folded, accomplish the unfolding procedure as covered in **MAIN ROTOR BLADE AND PYLON FOLDING,** Section IV. If the pylon and blades are unfolded, accomplish the following check to assure that the hydraulic unfolding operation is completed:

1. Rotor brake lever – OFF.
2. Blades unfolded indicator light – ON.
3. Flight position indicator light – ON.
4. Blade folding master switch – OFF.
5. Blade fold-unfold switch – OFF.
6. Safety valve switch – CLOSED.
7. Tail rotor coning switch (after pylon is unfolded) – FLT. POSIT. (close switch guard).
   Tail rotor coning warning lights should go out. Visually check that the coning ring is in the flight position.

NOSE DOOR AND RAMP CHECK.

If the nose doors and ramp are to be used, perform the following checks for proper operation:

CAUTION

Before lowering the ramp, be sure that the ground below the ramp is level and free from obstructions that would damage the ramp.

1. Control quadrant throttles – 1500 RPM.
2. Pilot's compartment ladder – FOLDED UP.
3. Nose door switch – CLOSED.
4. Nose door manual catch – RELEASE.
5. Nose door lock lever – UNLOCKED.
6. Nose door switch – OPEN, then OFF (when nose doors are open).
7. Ramp switch – DN, then OFF (when ramp is down).
8. Ramp switch – UP.
9. Ramp switch – OFF (when ramp up and locked indicator light comes on).
10. Nose door switch – CLOSE.
11. Nose door manual catch – LATCH and TIGHTEN.
12. Nose door lock lever – LOCKED.
    Check lock visually.
13. Nose door switch – OFF.

AUTOMATIC STABILIZATION HARD-OVER CHECK.

WARNING

The automatic stabilization equipment should not be operated with the main rotor blades in the folded position. The control rods and star may be damaged if the ASE is operated with blades folded.

Note

The following check must be performed with the blades in the flight position, the No. 1 engine running, clutch not engaged, and at least one generator operating, before the first flight after maintenance work has been done on the flight control system, the auxiliary servo system, or the automatic stabilization equipment. The check may be performed also whenever it is desired to check the motor circuit operation of the automatic stabilization system.

1. Left generator switch – ON.
2. Automatic stabilization circuit breaker and fuses (on overhead panel) – SET.
3. Cyclic control stick – CENTERED.
4. Tail rotor control pedals – CENTERED. Heels on pedal supports.
5. Collective pitch control – MINIMUM PITCH. Held firmly by the pilot.
6. ENGAGE button (on automatic stabilization control panel) – DEPRESS. Wait 2½ to 3½ minutes after external power is plugged in (or generator is operating) and depress; the green light in the button should come on.

Note

The rotor should not be turning for the check as described below.

7. ENGAGE button – STANDBY. Pull out (green light should go out).
8. Automatic stabilization adjustment box cover – OPEN.
9. Hard-over switch (on servo adjustment box) – LEFT.
10. Null indicator (on automatic stabilization control panel) – NEEDLE SHOULD SWING FULL LEFT.
11. Channel selector switch (on servo adjustment box) – TURN TO EACH CHANNEL (to make sure that all motors are hard over).

Note

When the hard-over switch is actuated, the automatic stabilization equipment introduces a steady full-authority signal in one direction to all four servo motors simultaneously, requiring a right pedal restraining force of 25 pounds.

12. Tail rotor control pedals – PUSH TO EXTREMES of travel in both directions. A force of up to 50 pounds will be required on the right pedal to move the pedals full travel in 24 seconds.
13. Cyclic control stick trim release switch – DEPRESS. Move stick longitudinally and laterally as far as possible at a rate not to exceed full travel in 1 second. (See note in step 14.)
14. Collective pitch controls – RAISE UP FOUR INCHES AND THEN DOWN (at a rate not to exceed full travel in 1 second). Back-off throttle when increasing pitch to prevent engine overspeed. A force of up to 10 pounds will be required to raise the collective pitch control.

Note

The maximum rates of control movement for the cyclic control stick right and aft and for the collective pitch control up (against the automatic stabilization signals) will be slower than in the opposite directions, but should not be less than a rate corresponding to full stick travel in approximately 1 second. Any resistance or seizing of the cyclic controls or excessive pedal or collective force indicates improper adjustment of the control linkage.

15. Hard-over switch – RIGHT. The collective pitch stick should be held during this check because the automatic stabilization equipment up signal will tend to drive the collective stick up and cause an engine overspeed. Although the force is limited to 10 pounds, the collective stick will move rapidly if it is not restrained.

CAUTION

The copilot should monitor the automatic stabilization servo shut-off switch for this check. A badly misadjusted servo, or bottomed open-loop spring could cause the collective pitch control to rise with a force of several hundred pounds. This action could be stopped only by turning off the servo or removing the hard over.

16. Channel selector switch – CHECK ALL CHANNELS.
17. Tail rotor control pedals – Move as in step 12.
18. Cyclic control stick – Move as in step 13.
19. Collective pitch control – Move as in step 14.

Note

When the hard-over switch is actuated to the RIGHT position, the same conditions noted above should be experienced except that the slower rates of control travel will be in the left, forward, and down directions. The 25-pound holding force and the 50-pound overriding force will be on the left tail rotor control pedal and the 10-pound holding force on the collective pitch stick will be down. The null indicator should swing full right.

WARNING

Do not operate the hard-over switch in flight since this introduces simultaneous hard-over signals on all four channels whether or not the automatic stabilization equipment is engaged. The results of using this switch in flight are quite severe and it should not be operated unadvisedly.

20. Hard-over switch – OP.
21. BAR ALT button – DEPRESS.
22. Collective pitch stick – RAISE SLIGHTLY.
23. Collective pitch control – RELEASE.
    Check that it returns to its original position.
24. BAR ALT button – PULL OUT.
25. Automatic stabilization engage button – DEPRESS (if automatic stabilization is desired for take-off).
    The automatic stabilization equipment possesses ample yaw authority to compensate for main rotor torque at take-offs and landings.

WARNING

During shipboard operations with the automatic stabilization equipment engaged, the feet should be on the tail rotor pedals at all times to disengage the yaw channel. Keeping the yaw channel disengaged is extremely important especially if the ship is turning as the turn will be resisted by the automatic stabilization directional channel and cause the helicopter to turn in relation to the deck of the ship.

Note

Any tendency for the tail rotor control pedals to creep should have corrected itself within 1 minute after ac and dc power is available, whether the automatic stabilization equipment is engaged or not. If this does not occur, the equipment may have been shut down before the servo motors were driven to null and this can be corrected by engaging the automatic stabilization equipment (after the 3-minute delay period).

CAUTION

If any erratic behavior in the control system is encountered when the automatic stabilization equipment is engaged, press the automatic stabilization release button on the cyclic stick grip.

26. CG trim control knob – CENTER.
27. Automatic stabilization servo adjustment box cover – CLOSED.

ROTOR CLUTCH ENGAGEMENT.

1. Manual rotor parking lock – CHECK DISENGAGED.
2. Tail rotor coning switch – FLT. POSIT.
   Tail rotor coning warning light out. Visually check tail rotor coning ring in the flight position.
3. Outside clearance – CHECK.
4. Rotor brake lever – OFF.

Note

The rotor clutch switches are inoperative unless the tail rotor is in the flight position, the hydraulic operations of main rotor blade unfolding are completed, and the rotor brake is off.

5. Collective pitch control – MINIMUM PITCH.
6. Mixture control levers – RICH.
7. Control quadrant throttles – APPROXIMATELY 1500 RPM.
8. Main gear box oil low pressure warning light – OFF.
9. Rotor clutch pump switches – ON.
   Check that the blades unfolded light goes on.
10. Control quadrant throttles – Advance slowly to 2100 rpm. Maintain 2100 rpm by increasing both throttles as the transmission system accelerates to 1400 apparent rotor rpm.
11. During clutch engagement, as the rotor speed accelerates to approximately 700 apparent rpm, move the cyclic control stick a slight amount in all directions and observe the tip-path plane of the main rotor for proper response of flight controls.

WARNING

If the flight controls do not respond correctly, do not complete the clutch engagement. Shut down by placing the clutch pump switches in the OFF position. This condition may be due to ice formation within the (flight control servo unit) boots (if boot is deteriorated and allows water to enter) rendering the servo pilot valves inoperative.

12. To engage the left clutch, reduce the left engine control quadrant throttle to allow the rpm of the left engine to drop below the apparent rotor rpm (engine tachometer pointer drops below rotor tachometer pointer). To complete the mechanical drive, advance the left engine control quadrant throttle sufficiently to line up the left engine and rotor pointers. Simultaneously, retard the right engine control quadrant throttle to prevent overspeeding of the engine as the mechanical drive of the left clutch will take over most of the transmission system load. The right engine control quadrant throttle should be retarded a sufficient amount to allow the rpm of the right engine to drop below the apparent rotor rpm (right engine tachometer pointer drops below rotor pointer), and then slowly advanced to complete the mechanical drive of the right clutch.

Note

Rotor speed should not be allowed to drop below approximately 1120 apparent rpm or the mechanical coupling of each clutch will not engage.

13. Control quadrant throttles – Adjust to 1500 rpm.
14. Rotor clutch pump switches – OFF. Rotor clutch pump warning lights will go out. Main gear box oil pressure warning light will go oft.

Note

- The right clutch may be engaged first, if desired, by reversing the above procedure.
- During cold weather, if the main gear box oil low pressure warning light is still on after clutch engagement, when the rotor clutch pump switches are turned off, the blades unfolded light will come on until the main gear box oil low pressure warning light goes off.

15. Check that all droop stop and anti-flapping restrainers have released. Droop stops should release at approximately 700 apparent rpm and anti-flapping restrainers should release at approximately 980 to 1120 apparent rotor rpm.
16. First and second stage servo shut-off switch – ON.

GROUND TESTS.

At 2000 rpm accomplish the following:

1. Battery switch – BATT.
2. External power – Disconnected or APU off. (Refer to **AUXILIARY POWER UNIT,** Section IV.)
3. Generator switches – ON. Check that generator warning lights go off.
4. Circuit breakers on overhead panel – SET.
5. Press-to-test all warning lights.
6. Ramp switch – UP. Check that green ramp up and locked warning light goes on. Return switch to OFF and green light should go off.
7. Radio master switch – ON.
8. Tune and test radio equipment.
9. Stabilizer positioning switch – HOVER.
10. Flight instrument power switch – NO. 1 INV. Check that flight instrument power warning light does not come on. Press light to test it. Place switch in NO. 2 INV. position. Warning light should not come on; return switch to NO. 1 INV.
11. Automatic stabilization servo switch – ON.
12. First and second stage servo shut-off switch – FIRST STAGE OFF.
    First stage servo hydraulic pressure gage should indicate a pressure drop and first stage hydraulic pressure warning light should come on. Check flight controls for proper operation on second stage servo hydraulic system.

Note

- Prior to making flight control checks, bleed air from the tandem servo hydraulic systems by full stroke slow cycling.
- In checking flight controls, move at a rate not to exceed full travel in 1 second.

CAUTION

Restrict control movements to a small range to prevent lifting or tilting the helicopter.

13. First and second stage servo shut-off switch – ON. Check first stage servo hydraulic pressure gage for a rise to normal operating range.
14. First and second stage servo shut-off switch – SECOND STAGE OFF.
    Second stage servo hydraulic pressure gage should indicate a pressure drop and the second stage servo hydraulic pressure warning light should come on. Check flight controls for proper operation on first stage servo hydraulic system.
15. First and second stage servo shut-off switch – ON. Both servo hydraulic pressure gages should indicate normal pressure and both servo hydraulic pressure warning lights should be off.
16. Automatic stabilization servo shut-off switch – OFF.

Note

- Check that ASE is disengaged before turning off the automatic stabilization servo shut-off switch.

- Do not apply pressure to the tail rotor control pedals with the automatic stabilization servo shut-off switch in the OFF position.

17. Twist-grip throttle — CHECK THAT RESISTANCE TO ROTATION MAY BE OVERCOME.
18. First and second stage servo shut-off switch — FIRST STAGE OFF.
First stage servo hydraulic pressure gage should indicate a pressure drop and the first stage hydraulic pressure warning light should come on.
19. First and second stage servo shut-off switch — ON.
Check first stage servo hydraulic pressure gage for rise to normal operating range.
20. First and second stage servo shut-off switch — SECOND STAGE OFF.
Second stage servo hydraulic pressure gage should indicate a pressure drop and the first stage hydraulic pressure warning light should come on.
21. First and second stage servo shut-off switch — ON.
Both servo hydraulic pressure gages should indicate normal pressure and both servo hydraulic pressure warning lights should be off.
22. Automatic stabilization servo shut-off switch — ON.
23. Fuel gage test switches — Push to test fuel quantity gages for proper operation.
24. Volt-ammeters — 28.5 volts at 1500 rpm. Amperes — Note increase when radio equipment is turned ON.
25. Gyro magnetic compass function switch — FREE or SLAVED, as desired.

AUTOMATIC STABILIZATION EQUIPMENT CHECK.

Under normal conditions it is only necessary to set the cg trim and engage the stabilization equipment. For night or instrument flight, the following checks may be performed on the ground before take-off or while taxiing. If these checks are being performed while not taxiing, the tail wheel should be unlocked and tie-downs removed.

1. Move the cyclic control stick fore-and-aft and note that the null indicator follows the movements of the stick.
2. Directional control may be checked by either of the following methods: When taxiing is not necessary, remove feet from tail rotor pedals and increase power. The helicopter should not change direction. While taxiing, remove feet from tail rotor pedals and slowly turn the yaw trim knob in either direction. Slight changes should be noted in the heading of the helicopter. Adjust the yaw trim knob to guide the helicopter in the desired direction with feet removed from pedals.

Note

The ability of the stabilization equipment to hold a heading while taxiing or to turn when performing the directional control checks will depend on apparent rotor rpm, wind velocity, runway surface, etc. A tendency to overshoot, especially after commanding a sizeable heading change with the yaw trim knob, does not necessarily indicate stabilization equipment malfunction and may be easily suppressed by the pilot's assistance on the tail rotor pedals.

TAXIING.

A helicopter take-off can often be accomplished from the ramp or parking area after a turn into the wind. Caution should be observed when there are light planes within the immediate area as they may be upset by the turbulence caused by the rotors. A short low speed flight at low altitude may be more practical than taxiing; however, if necessary to taxi, proceed as follows:

CAUTION

Before taxiing, retract ramp, close nose doors, check that external power is disconnected, and check that the generator switches are on.

1. All tie-downs and chocks removed; external power — Disconnected.
2. Parking brake handle — OFF.
3. Tail wheel lock handle — UNLOCKED.
4. Collective pitch control — MINIMUM PITCH.
5. Twist-grip throttle indicator — 40 degrees.
6. Control quadrant throttles — 2300 RPM.
Equalize manifold pressures. If further power changes are required, control both engines simultaneously with the twist-grip throttle.

Note

Maintain a minimum of 2300 rpm while taxiing so that an immediate take-off can be accomplished if a crosswind should tilt the helicopter or ground resonance should develop. When taxiing crosswind, hold cyclic control stick slightly into the wind.

Note

If the landing gear is in the stowage position, the oleo struts must be fully extended prior to taxiing.

7. Cyclic control stick – FORWARD.
8. Collective pitch control – INCREASE SLIGHTLY. Increase collective pitch until forward motion is obtained.

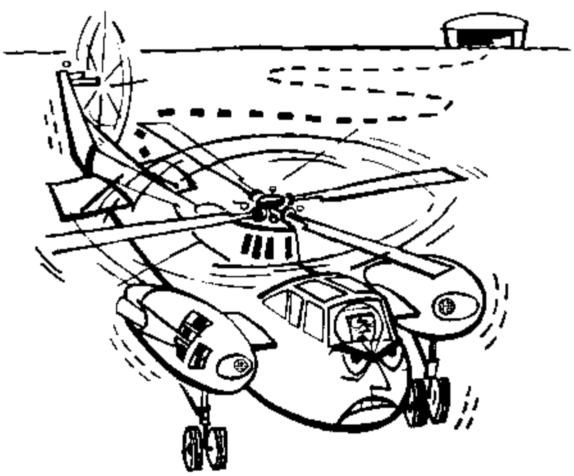
CAUTION

Use the minimum collective pitch required for forward motion. This should be enough to prevent the main rotor blades from hitting the droop stops with the cyclic control stick forward.

9. Regulate taxiing speed with fore-and-aft movement of the cyclic control stick.
10. Maintain directional control by use of the tail rotor control pedals. Use the wheel brakes if necessary.

CAUTION

Due to the full swiveling tail wheel and the thrust developed by the tail rotor, the helicopter may have a tendency to "oversteer" while taxiing. Use tail rotor control pedals cautiously, especially when executing turns.



GO EASY WITH THE TAIL ROTOR CONTROL PEDALS AT TAKE-OFF AND LANDING! THE STRONG TAIL ROTOR AND CASTERING TAIL WHEEL CAN REALLY SWING THE AFT END AROUND!

11. Check that directional indicators register taxi turns.
12. Before taxiing is stopped, be sure that the swiveling tail wheel is in the centered position so that it may be locked before take-off.
13. Stop the helicopter by heading into the wind or by applying the wheel brakes.

CROSSWIND EFFECTS.

When taxiing crosswind, the helicopter will have a tendency to weathercock into the wind. This condition is a result of the wind striking the tail cone and pylon and can be minimized by locking the tail wheel when not making turns and by holding the cyclic control stick into the wind. The latter also prevents excessive coning of the main rotor blades which would create a tendency for the helicopter to tip over during strong winds. Weathercocking can be corrected by proper application of tail rotor control pedals and wheel brakes.

BEFORE TAKE-OFF.

(Refer to figure A–4.)

PREFLIGHT ENGINE CHECK.

1. Fuel booster pump switches – EMER.
2. Fuel cross-feed switch – CLOSE.
3. Carburetor air levers – COLD.
4. Mixture control levers – RICH.
5. Fuel shut-off switches – ON.
6. Ignition system check – Retard right control quadrant throttle to approximately 1800 engine rpm and advance left control quadrant throttle and collective pitch to produce 2200 rpm at field barometric pressure on manifold pressure gage. Accomplish L and R magneto checks for the left engine. Maximum drop-off is 100 with a maximum differential between L and R of 40.

Note

The reduction of the right engine rpm to 1800 indicates that the freewheeling unit of the right clutch is operating correctly.

7. Allowing the collective pitch to remain in position, advance the right control quadrant throttle and retard the left control quadrant throttle so that the right engine produces 2200 rpm and the left engine 1800 rpm. Repeat the magneto check for the right engine and the freewheeling unit check of the left clutch.

CAUTION

Engine roughness, such as that caused by ignition malfunction, is transmitted directly to the rotors and causes abnormal vibration in the form of twitching or lashing of the helicopter structure. It stresses the transmission and rotor and may cause damage or lead to failure of these components.

WARNING

If the engine tachometer pointer does not drop to approximately 1800 rpm when the control quadrant throttle is retarded, the freewheeling unit of the clutch is malfunctioning. Do not take off as autorotation and possibly single-engine flight would be impossible with the free-wheeling unit of the failed engine malfunctioning.

8. Control quadrant throttles – Hold twist-grip throttle at 40 degrees and equalize the manifold pressures of both engines at 2500 rpm with control quadrant throttles. Accomplish subsequent power changes with the twist-grip throttles.
9. Check all engine and transmission temperature and pressure gages for readings within the normal operating ranges at 2500 engine rpm.

PREFLIGHT AIRCRAFT CHECK.

1. Cabin heater switch – As required.
2. Pitot heater switch – ON (if icing conditions prevail).
3. Flight instrument power switch – NO. 1 INV.
4. Hydraulic pressure gages – Check first and second stage servo and utility system hydraulic pressure gages for reading within normal range.
5. Stabilizer positioning switch – As desired.
6. Check all flight controls for proper operation.

CAUTION

Restrict control movements to a small range to prevent lifting or tilting the helicopter.

7. Tail wheel lock handle – AS REQUIRED.

Note

When the tail wheel lock handle is placed in the LOCKED position, the tail wheel will not actually be locked unless it is in the centered position.

8. Make prepare-for-take-off announcement over the interphone if personnel are carried in the cabin. Announcement should be confirmed from the cabin.
9. Parking brake handle – UNLOCKED.
10. Throttle authority – RECHECK.

TAKE-OFF (NORMAL VERTICAL).

The normal vertical take-off is the most common type of take-off since it may be used when operating at normal gross weights and at moderate altitudes. Gross weight, altitude, and temperature conditions at which vertical take-offs can be accomplished may be determined by reference to the Take-Of Distances Charts (figures A–10, A–11, and A–12) in Appendix I. Where zeros appear in the column of figures A–10, A–11, and A–12, headed IAS KNOTS, vertical take-offs can be accomplished. This type of take-off provides a high safety factor as the helicopter is lifted vertically from the ground to hover at approximately 10 feet, and the flight controls and engines may be checked for normal operation before beginning to climb.

1. Collective pitch control – MINIMUM PITCH.
2. Twist-grip throttle – ADVANCE TO TAKE-OFF RPM.
3. Collective pitch control – Increase pitch steadily as helicopter leaves the ground. Maintain take-off rpm by rotating twist-grip throttle.

Note

During take-off, the copilot should monitor the control quadrant throttles to keep the manifold pressures equalized.

4. Prevent the helicopter from swerving by use of the tail rotor pedals if automatic stabilization is not used. Apply left pedal to counteract increasing main rotor torque. Use pedals cautiously to avoid scuffing tires sideways at take-off.
5. Rise vertically to approximately 10 feet, maintaining level attitude with cyclic control stick.

Note

Refer to **ENGINE FAILURE,** Section III, for procedure to be followed in the event of engine failure on take-off.

WARNING

Carbon monoxide may increase rapidly when hovering with engine exhaust to the windward with cockpit windows either open or closed.

TAKE-OFF (MAXIMUM PERFORMANCE).

Maximum performance take-offs enable the helicopter to take-off at high gross weights or high altitude where hovering performance is marginal. This type of take-off should be accomplished when hovering is possible in ground effect, but not possible out of ground effect. The primary objective of a maximum performance take-off is to rise vertically until airborne and then immediately accelerate into forward flight without hovering, thereby gaining the additional lift developed in forward flight before establishing a climb. Certain factors must be considered to determine that take-off and air acceleration can be accomplished. Each of the following factors are critical and must be carefully studied prior to attempting this type take-off.

1. Gross weight.
2. Density, altitude, and outside air temperature.
3. Wind velocity, direction, and constancy.
4. Size of take-off area and area available for air acceleration and climb-out.

Note

When take-offs at high gross weight or under high altitude conditions are necessary, optimum performance may be obtained by placing the stabilizer in the hover position.

Gross weight, temperature, and altitude conditions requiring maximum performance take-offs are shown on the Take-Off Distances Charts (figures A–10, A–11, and A–12) in Appendix I. When values, other than zero, appear in the column of figures A–10, A–11, and A–12, headed IAS KNOTS, maximum performance take-offs are necessary. After a vertical take-off, the helicopter should be accelerated in level flight to the values shown in the IAS KNOTS column before beginning to climb. Distances required, at zero wind velocity, to attain these airspeeds, are shown in the column headed ACCEL. DIST.

1. Head the helicopter into the wind.
2. Collective pitch control – MINIMUM PITCH.
3. Twist-grip throttle – ADVANCE TO TAKE-OFF RPM.

CAUTION

On helicopters prior to BuNos 145875, unless modified by ASC No. 44A, rapid increase in pitch of main rotor blades may result in excessive torque in main rotor drive. During take-off, allow a minimum of 2 seconds to move the collective pitch stick from low to high pitch.

4. Collective pitch control – Increase collective pitch steadily until the helicopter becomes airborne. When operating with high values or high combinations of gross weight, altitude, and temperature, it may be necessary to increase collective pitch to such an extent that maximum take-off manifold pressure is obtained.
5. As soon as the helicopter rises to a sufficient height to prevent striking the landing gear wheels on the ground, ease the cyclic control stick forward, increase collective pitch until maximum take-off manifold pressure is obtained, and accelerate forward in level flight.

CAUTION

If maximum take-off manifold pressure is used to become airborne, a slight settling may be encountered when the cyclic control stick is eased forward. This is due to the tilting of the tip-path plane of the main rotor blades and a decrease in ground effect as the helicopter moves forward. Care should be taken to prevent striking the landing gear wheels on the ground should settling occur.

6. Tail rotor control pedals – Hold the helicopter on a steady heading into the wind.
7. Accelerate in level flight until airspeed increases to the IAS value shown on figures A–10, A–11, and A–12 for the gross weight and altitude conditions at the time of take-off.
8. When proper IAS value is obtained, begin to climb. After clearing obstructions, increase airspeed to approximately 70 knots for best rate of climb and for single-engine protection.

AFTER TAKE-OFF.

After a normal vertical take-off, check the following unless hovering performance is marginal and an air acceleration is necessary.

1. Flight controls for normal operation.
2. All instruments for operation within normal range.
3. Automatic stabilization attitude control check. Establish the desired hovering attitude. If the null indicator is operating about a position appreciably off center, the cg trim knob should be re-adjusted. Move the knob in the direction that is necessary to move the null indicator pointer to the center. This will require repositioning of the pilot's cyclic stick to maintain the hover, but will result in a relocation of the average pointer position in the desired direction. Utilize the cyclic control stick trim system to position the cyclic stick. The helicopter should stabilize at the desired attitude without additional cyclic control being applied by the pilot.
4. Automatic stabilization directional control check. Remove feet from pedals; no change in heading should occur. Make a definite power change; no change in heading should occur.

CLIMB.

After a normal vertical take-off or after clearing obstructions in a maximum performance take-off, accelerate forward in a climb until the best rate-of-climb airspeed is obtained. The airspeed that will produce the best rate of climb for various gross weights and altitudes may be determined by reference to the Climb Charts, figures A—13, A—14, and A—15. While establishing a climb, accomplish the following:

CAUTION

Vertical climbs to clear ground obstructions can be accomplished after normal vertical take-offs; however, for normal operation, airspeeds between 0 and 40 knots should be avoided at low altitudes as a safe autorotative landing would be difficult to perform should engine failure occur at low altitude and low airspeed.

1. Power settings — Adjust in accordance with figures A—13, A—14, and A—15.
2. Mixture control levers — NORMAL.
3. Fuel booster pump switches — NORM.
4. Carburetor air levers — As required.
5. Manifold pressures — Equalize, if necessary, by control quadrant throttles.
6. Stabilizer positioning switch — FWD. FLIGHT.
7. Cyclic control stick trim release switch or beeper trim switch — trim cyclic control stick to desired position.
8. Landing gear actuating lever — UP after safe single-engine airspeed is attained. Landing gear position indicators should indicate both wheels up and locked after approximately 5 seconds.

HOVERING.

Maximum performance for hovering out of ground effect may be accomplished by placing the stabilizer positioning switch in the HOVER position.

DURING FLIGHT.

1. Refer to figures A—17, A—18, and A—19 for cruising power settings.

Note

When operating with approximately 30 inches of manifold pressure and NORMAL mixture, engine surging in excess of 100 rpm can be encountered. This surging can be eliminated either by moving the mixture control lever to RICH or by changing the power setting.

2. Cyclic control stick trim release switch or beeper trim switch — Adjust cyclic control stick for cruise.
3. Automatic stabilization BAR ALT button — Depress, after leveling off, stabilizing airspeed and power; adjust friction on the collective pitch control to allow free motion using only enough friction to suppress any hunting tendencies of the collective stick in gusty weather.
4. Equalize manifold pressures as required with control quadrant throttles.

Note

When equalizing manifold pressures, use one overhead throttle only during the flight to preclude any possibility of the quadrant throttles getting out of adjustment with the twist-grip throttles.

FLIGHT CHARACTERISTICS.

Information on the flight characteristics of the helicopter is covered in section VI.

SYSTEMS OPERATION.

Information regarding the operation of the various systems is covered in sections IV and VII.

TURNS USING AUTOMATIC STABILIZATION.

Turns using the automatic stabilization equipment may be made in any of two ways: (1) By actuating the tail rotor control pedals. After the turn is completed and the helicopter is on the desired course, the feet should be removed from the pedals and the helicopter will maintain the new heading. (2) By use of the yaw trim knob. For turns while hovering, rotate the knob left or right slowly and smoothly to produce the desired turn. With forward speed, this control will be very convenient for small turns of 1 to 10 degrees. Large turns can be made but will cause the helicopter to skid unless they are made very slowly or the pilot banks the helicopter while the knob is rotated to prevent skidding.

PRELANDING CHECK.

1. Carburetor air levers — COLD.
2. Mixture control levers — RICH.
3. Fuel shut-off switches — ON.
4. Fuel booster pump switches — EMER.
5. Cabin heater switch — OFF.
6. Automatic stabilization BAR ALT button — Pull to disengage. Check null indicator centered.
7. Parking brake handle — As desired.
8. Tail wheel lock handle — As desired (LOCKED for shipboard landings).
9. Landing gear lever — DN.
   Check landing gear position indicators. Warning light in lever knob should come on, then go out when landing gear is down and locked. Landing gear should extend in approximately 5 seconds.
10. Twist-grip throttle indicator — 40 degrees. Refer to **QUADRANT THROTTLE AND TWIST-GRIP THROTTLE OPERATION,** Section VII.

Note

If the landing gear fails to lower, or if the warning light in the knob fails to go out, refer to **LANDING GEAR FAILURE,** Section III.

11. Twist-grip throttles – 2600 RPM.
12. Approach speed – 60 to 70 KNOTS.
13. Stabilizer positioning switch – HOVER.
14. Make prepare-for-landing announcement over interphone if personnel are carried in the cabin.

CAUTION

During all approaches to a landing or hover, the control quadrant throttles are to be monitored by the copilot to prevent throttle movement in either direction.

LANDING (POWER-ON VERTICAL).

(Refer to figures A–16 and 2–2.)

1. Perform prelanding check.
2. Gradually decrease airspeed and rate of descent on final approach leg by moving the cyclic control stick aft and increasing collective pitch.
3. Engine rpm – Maintain 2600 rpm by increasing twist-grip throttle as airspeed decreases.
4. Attain a hovering attitude at approximately 10 feet altitude.
5. Maintain a level attitude and sufficient altitude to prevent the wheels from scuffing the ground before final touchdown. Maintain a constant heading and eliminate all side drift by coordinated use of the cyclic control stick and tail rotor pedals. The automatic stabilization system, when engaged, will provide the necessary tail rotor corrections to maintain a constant heading if the pilot and copilot keep their feet off the tail rotor control pedals. When collective pitch is decreased, torque developed by the main rotor will rapidly decrease and tend to turn the helicopter to the left, unless right tail rotor control pedal is depressed to decrease tail rotor thrust. If ground contact can be made with a slight forward roll, less right tail rotor control pedal correction will be required. The castering effect of the tail wheel in forward motion will tend to stabilize the helicopter in yaw.
6. Reduce collective pitch for final vertical descent, with 0 to 10 knots forward speed if space permits.
7. As the wheels contact the ground, steadily reduce collective pitch to almost minimum to prevent the possibility of a gust of wind tipping the helicopter and to prevent ground resonance. Use tail rotor control pedals cautiously to avoid swerving.

Note

Maintain 2600 engine rpm for a quick take-off until it is certain that the wheels are on solid ground.

8. Engine rpm – Reduce to 2300 rpm for taxiing or postflight engine check.

WAVE-OFF.

1. Increase power to take-off power if necessary to clear obstructions.
2. Attain forward speed and establish a climb.
3. Reduce power to maintain climb to desired altitude.

AFTER LANDING.

1. Taxi to head the helicopter into the wind with the tail wheel centered before engine shutdown.
2. Mixture control levers – RICH.
3. Tail wheel lock handle – LOCKED.
4. Parking brake handle – ON.
5. Collective pitch control – MINIMUM PITCH.
6. Carburetor air levers – COLD.

POSTFLIGHT ENGINE CHECK.

The postflight engine check should be accomplished after the last flight of the day. Any unsatisfactory conditions should be reported on the appropriate forms.

1. Mixture control levers – RICH.
2. Check all engine and transmission instruments for operation within the normal operating range at 2600 rpm.
3. Adjust twist-grip throttle to approximately 40 degrees on the twist-grip throttle position indicator. Use one or both quadrant throttles for the following checks.
4. Ignition system check – Retard right control quadrant throttle to approximately 1800 rpm and advance left control quadrant and collective pitch to produce 2200 rpm at a manifold pressure equal to field barometric pressure. Accomplish L and R magneto check on left engine. Drop-off on either L or R position should not exceed 100 rpm. Difference in drop between L and R should not exceed 40 rpm.

Note

The reduction of the right engine rpm to 1800 indicates that the freewheeling unit of the right clutch is operating correctly.

5. Repeat this check with the left control quadrant closed to accomplish the ignition system check on the right engine and the freewheeling unit check of the left clutch.

WARNING

If the engine tachometer pointers do not

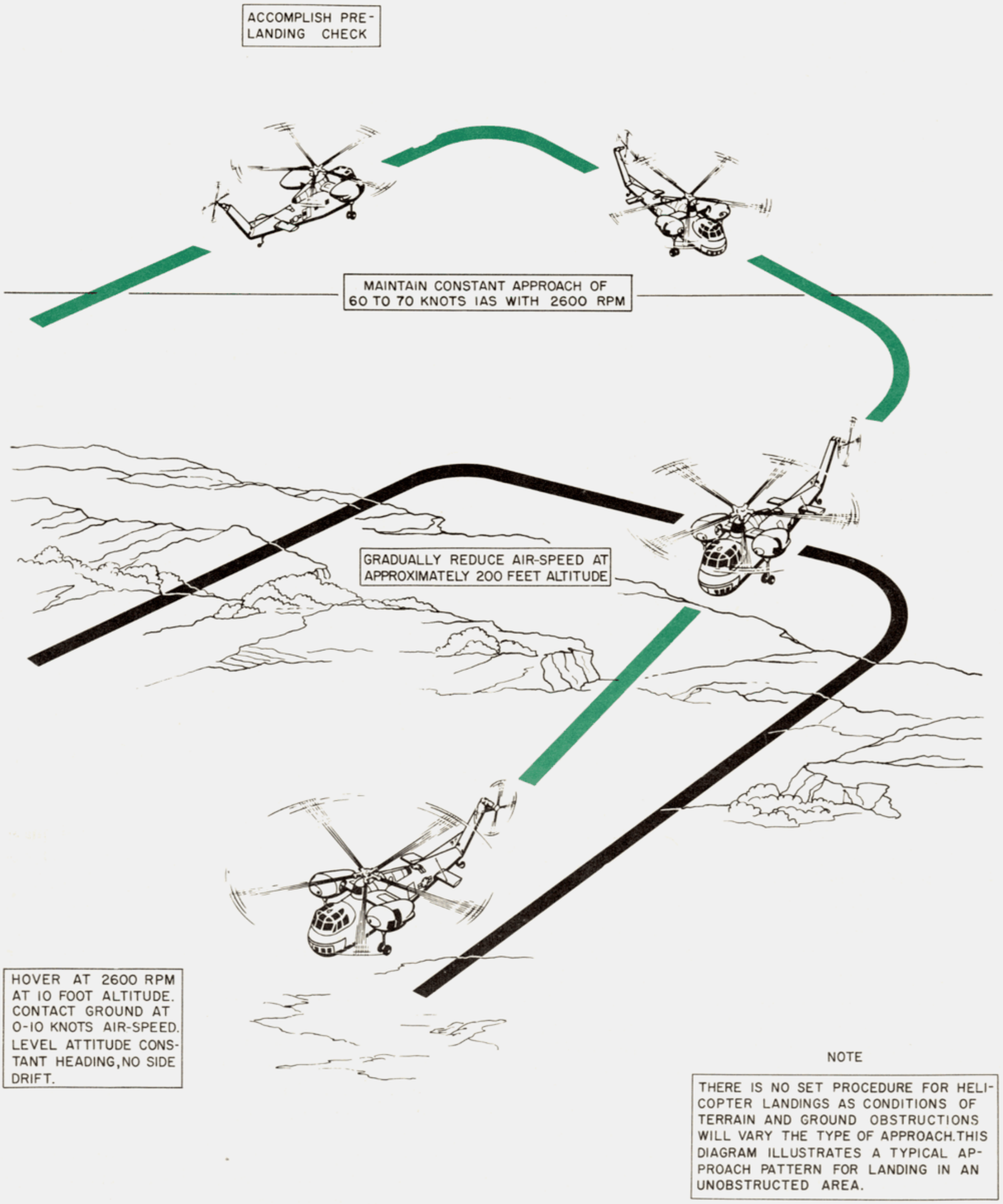


Figure 2–2. Power-On Landing

drop to approximately 1800 rpm when the control quadrant throttle is closed, the freewheeling unit of the clutch is malfunctioning.

6. Automatic stabilization engage button – Pull out.
7. Disengage rotor by reducing engine speed to 1000 rpm with collective pitch control in minimum position.
8. First and second stage servo shut-off switch – FIRST STAGE OFF.
   As rotor head slows and before the blades begin to droop.
9. Idle speed check – Engine should idle at 1000 rpm.
10. Droop stops– In place. (Flyweights retracted.)

Note

Droop stop malfunction can be detected by observing the tip-path of the main rotor blades or the position of the flyweights. A low blade or an extended flyweight indicates that the droop stop is not in place. When this condition is noted, rotor rpm should be increased and the cyclic control stick actuated slightly to lift the blade and allow the flyweight to retract as rotor rpm decreases permitting droop stop actuation. When a droop stop cannot be actuated (flyweight remains extended), the cyclic control stick should be held in approximately the neutral position as rotor rpm decreases.

11. Rotor brake lever – Apply intermittently below 980 apparent rotor rpm to stop rotors, then leave on (lever forward).
12. Tail rotor coning switch – CONED after rotors stop turning.

AUXILIARY GROUND EQUIPMENT.

1. To fold blades and pylon, refer to **MAIN ROTOR BLADE AND PYLON FOLDING,** Section IV.
2. To open nose doors and lower ramp, refer to **NOSE DOORS AND RAMP,** Section IV.

Since the above operations require utility hydraulic power, they should be performed with the left engine or both engines running.

STOPPING OF ENGINES.

1. Throttles – CLOSED.
2. Fuel booster pump switches – OFF.
3. Idle engines at 1000 rpm with the mixture control levers in the NORMAL position until cylinder head temperature is less than 200°C.
4. Idle mixture check – Move both mixture control levers slowly to the IDLE CUT-OFF position. A speed increase of 0 to 20 rpm before the engine stops indicates proper idle mixture setting.
5. Fuel shut-off switches – OFF.
6. Ignition switches – OFF.
7. Fuel feed valve check – After the engines stop, re-open mixture controls to NORMAL position.
8. Fuel pressure gage – Pressure should fall off to 8 to 10 psi and gradually decrease. On BuNos 140314 and 140316 through 140319, the fall-off rate will be slightly higher than in the others since the fuel shut-off valve has a bleed hole in it. A leaking fuel valve can only be detected by an increase in the normal pressure fall-off rate.

BEFORE LEAVING THE HELICOPTER.

1. All switches and rheostats on overhead switch panel – OFF. Except cyclic control stick trim master switch – ON.
2. Tail wheel lock handle – LOCK.
3. Rotor brake lever – ON (forward).
4. First and second stage servo shut-off switch – FIRST STAGE OFF.
5. Radios – OFF.
6. Parking brake handle – SET (brake on).
7. Main rotor blade folding master switch – OFF.
8. Tail rotor coning switch – CONED.
9. Parking light switch (forward of cabin door) – PARKING LIGHT (if required) or OFF.

Note

A notation of any contact with salt water spray should be made on the appropriate forms in order that the ground crew can take preventive action against corrosion.

10. Wheel chocks in place.

SECTION III
EMERGENCY PROCEDURES

ENGINE FAILURE.

FAILURE OF ONE ENGINE.

Immediately upon failure of one engine, increase the throttle on the remaining engine, reducing or increasing collective pitch to maintain rotor rpm and pull the fire emergency handle for the failed engine. The mixture control lever for the operative engine should be placed at RICH and an airspeed of 65 to 75 knots established. A landing should be made as soon as practicable. Refer to SINGLE ENGINE LANDING in this section. If a restart is considered safe, proceed as directed under RESTARTING ENGINE IN FLIGHT in this section. If not, refer to ENGINE SHUTDOWN IN FLIGHT, in this section.

Failure of the left engine will result in loss of the utility hydraulic system which powers:

- Twist-Grip Throttle Servo
- Tail Rotor Pedal Damper
- Auxiliary Servo Unit
- Landing Gear
- Wheel Brakes
- Nose Door and Ramp
- Stabilizer
- Blade Folding

Failure of the right engine will result in loss of the second stage servo unit.

RESTARTING ENGINE IN FLIGHT.

1. Control quadrant throttle for inoperative engine – Closed and locked.

CAUTION

The required power (twist-grip throttle and application of collective pitch) for the operating engine must be established before the control quadrant throttle for the failed engine is closed and locked. Power changes will be difficult, if not impossible, with one quadrant throttle closed and locked.

2. Mixture control lever – IDLE CUT-OFF.
3. Fuel shut-off switch – ON.
4. Fuel booster pump switch – EMER.
5. Starter switch – DEPRESS.
6. Ignition switch – BOTH.
7. Primer switch – After the starter has been engaged for 15 seconds, while continuing to crank, intermittently prime with short rapid pulses if engine is warm, longer sustained pulses if engine is cold.
8. Mixture control lever – Continue intermittent priming until engine operates smoothly at 1000 rpm, then move the mixture control lever to NORMAL.
9. After the engine starts, check for a rise in oil pressure.

Note

If oil pressure does not register on gage almost immediately, stop engine.

10. If time and altitude permit, warm up engine in a normal manner.
11. Advance throttle slowly to driving speed. As control quadrant throttle is advanced, back off on the twist-grip throttle to prevent overspeeding of the engines.

Note

It is not necessary to re-engage the hydro-mechanical clutch, as the mechanical coupling will remain engaged and in the freewheeling position because of the high rpm.

ENGINE SHUTDOWN IN FLIGHT.

1. Engine fire emergency handle – PULL.
2. Ignition switch – OFF.
3. Fuel booster pump switch – OFF.
4. Mixture control lever – IDLE CUT-OFF.
5. Fuel shut-off switch – OFF.
6. Quadrant throttles – Maintain relationship of both throttles.

Note

When the left engine fails, rendering the twist-grip throttle servo inoperative, placing the left engine control quadrant throttle in the CLOSE position will increase the existing friction loads in the twist-grip throttle system. It is therefore recommended that whenever conducting an engine shutdown in flight, the quadrant throttles should be maintained in the same relationship that existed prior to the necessity for an engine shutdown.

ENGINE FAILURE DURING TAKE-OFF OR WHILE HOVERING AT LOW ALTITUDE.

1. Collective pitch – Reduce momentarily to retain rotor rpm.
2. Throttle and collective pitch – Increase to take-off power.
3. If engine failure occurs while hovering, hold the helicopter on a level attitude as it settles to the ground.
4. If at a sufficient height, attempt to gain as much forward speed as possible (up to 70 knots) and accomplish a partial flare to cushion the landing.

SINGLE ENGINE LANDING.

Select a landing site with a smooth surface to accomplish a landing.

1. Crew – ALERTED.
2. Carburetor air lever – COLD.
3. Mixture control lever – RICH.
4. Parking brake handle – UNLOCKED.
5. Tail wheel lock handle – AS DESIRED.
6. Landing gear actuating lever – DN.

Note

If the left engine has failed, the utility hydraulic system will be inoperative. Proceed in accordance with **UTILITY HYDRAULIC SYSTEM FAILURE** in this section.

7. Engine speed – 2600 RPM.
8. Approach airspeed – 65 TO 75 KNOTS.
9. Stabilizer positioning switch – HOVER.
10. Accomplish a normal approach at 65 to 75 knots.

CAUTION

A wave-off below 60 knots of airspeed would be marginal. If necessary to go-around, apply take-off power and establish a climb at between 60 and 70 knots.

11. Begin a partial flare at approximately 100 feet to reduce airspeed and rate of descent.
12. Level off with approximately 30 knots airspeed using full power if necessary.

Note

For landing on an unprepared surface, a zero ground speed is desired.

13. Ground contact should be in a level attitude. Gradually decrease airspeed and increase power.
14. As the wheels contact the ground, steadily reduce collective pitch to minimum.

FAILURE OF BOTH ENGINES.

DURING TAKE-OFF OR WHILE HOVERING AT LOW ALTITUDE.

Maintain a level attitude and increase collective pitch to maximum before ground contact. If altitude permits, attempt to gain as much forward airspeed as possible (up to 70 knots) and accomplish a partial flare to cushion the landing.

DURING FLIGHT (AUTOROTATIVE LANDING). *(Refer to figure 3–1.)*

1. Collective pitch – Reduce immediately to retain rotor rpm.
2. Engine fire emergency handles – Pull.
3. Establish a 60 to 70 knot glide.
4. Maintain apparent rotor rpm between 2660 and 2800 with collective pitch.
5. If altitude permits, lower the landing gear, using the emergency hydraulic system.

Note

Lowering the landing gear with the emergency system will take approximately 2 minutes.

6. Fuel booster pump switches – OFF.
7. Ignition switches – OFF.
8. Fuel shut-off switches – OFF.
9. Alert crew.
10. Battery switch – OFF.
11. At approximately 100 feet above the ground, begin flare to reduce rate of descent and airspeed. Reduce airspeed to between 30 and 40 knots in flare.

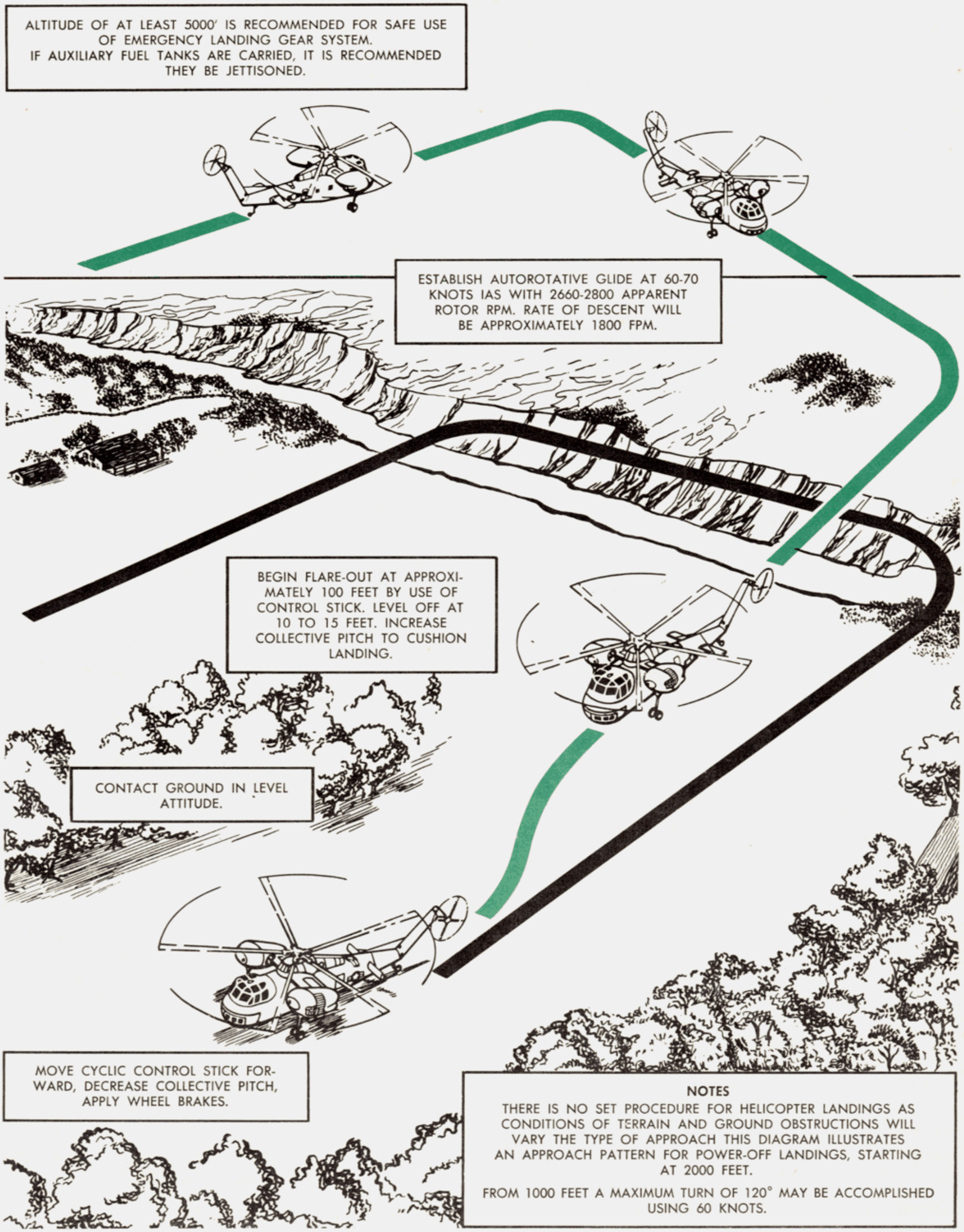


Figure 3—1. Autorotative Landing

12. Level off at approximately 10 to 15 feet and increase collective pitch to cushion the landing.
13. Contact the ground in a level attitude.
14. When the wheels contact the ground, decrease collective pitch to minimum, center the cyclic control stick, and apply the rotor and wheel brakes.

FIRE.

ENGINE FIRE WHILE STARTING.

Continue with the starting procedure if the fire is confined to the induction or exhaust system. If the engine does not start or the fire is not blown out, proceed as follows:

1. Fire emergency handle – Pull (to shut off fuel and oil and to select nacelle for fire extinguisher).
2. Control quadrant throttle – CLOSED.
3. Mixture control lever – IDLE CUT-OFF.
4. Ignition switch – OFF.
5. Fuel booster pump switch – OFF.
6. Fuel shut-off switch – OFF.
7. If ground crew cannot extinguish the fire, warn them to stand clear and turn on appropriate engine fire extinguisher switch.

WARNING

Vapors from the fire extinguisher agent, Bromotrifluoromethane, while not poisonous, can cause asphyxiation through reduction of oxygen, especially in confined spaces. The liquid can cause low temperature burns when in contact with the skin. All personnel should stand clear and in the open air. If the helicopter is on the ground, the cabin should be vacated as a precautionary measure.

Note

Whenever the engine fire extinguishing system has been used, replace the discharged spherical container, purge all lines, and thoroughly clean and ventilate all contaminated areas as soon as possible, as the action of fire extinguishing agent on metal is corrosive.

ENGINE FIRE AFTER STARTING.

If an engine fire develops after starting, advance the throttle to attempt to blow out the fire. If the fire continues to burn, proceed as instructed under ENGINE FIRE WHILE STARTING in this section.

ENGINE FIRE IN FLIGHT.

1. Engine fire emergency handle – Pull out.
2. Engine fire extinguisher switch – Select the appropriate switch and place at ON. The extinguisher is used for only one engine and one time.
3. Proceed as in ENGINE FAILURE and ENGINE SHUTDOWN IN FLIGHT in this section.

Note

The prime consideration in all fire-in-flight emergencies is to land immediately.

FUSELAGE FIRE.

1. Pilot's compartment sliding windows – CLOSED.
2. Cabin doors – CLOSED.
3. Ventilating fan switch – OFF (if installed).
4. Use the portable fire extinguisher (8, figure 3–4).

Note

Land as soon as possible. Determine cause of fire before continuing flight.

ELECTRICAL FIRE.

Possibilities of electrical fires are slight because the generators are prevented from generating excessive voltage by the field control relays and each electrical circuit is protected from overload by circuit breakers. However, in event of an electrical fire, attempt to isolate the circuits affected by pulling circuit breakers. If fire persists, land as soon as possible; while in flight accomplish the following:

1. Attempt to isolate circuits affected by pulling circuit breakers.
2. Prepare to land if fire persists.
3. Generator switches – OFF.
4. Battery switch – OFF (after lowering landing gear).
5. Use the portable fire extinguisher (8, figure 3–4).

SMOKE ELIMINATION.

After a fire is extinguished, smoke may be eliminated by opening the pilot's compartment sliding windows. Open the door in the cabin aft bulkhead. Do not push out windows or open the cargo door while the helicopter is in flight, because of the possibility of their being carried into the tail rotor blades by the airstream.

EMERGENCY ENTRANCES.

Emergency entrances (figure 3–2) to the pilot's compartment are the jettisonable side windows. Emergency entrances to the cabin are the nose door emergency panels, the upper panel of the passenger door, the emergency hatch on the left side of the fuselage, and the cabin windows. All emergency entrance handles are marked as described in **EMERGENCY EXITS,** Section I.

DITCHING.

The helicopter is an ideal aircraft to ditch since it is capable of contacting the water with little or no forward speed. However, it has poor floating tendencies. By virtue of its versatility, most landings may be safely accomplished either by briefing the crew or by ditching drills prior to all overwater flights.

PLANNED DITCHING — POWER ON.

In the event of anticipated fuel shortage during overwater flights, or for any other unforeseen reasons when ditching the helicopter is imminent but not immediate, much can be done to further protect personnel by having a planned ditching procedure.

PLANNED DITCHING PROCEDURE.

1. Brief crew prior to take-off.
2. Immediately alert crew over the interphone when a ditching appears inevitable.
3. Order crew to prepare all sea survival gear for aerial drop (gear should be secured in daisy chain fashion).
4. Transmit distress message.
5. Life vest — Check security.
6. Hover helicopter, jettison sliding windows in pilot's compartment, and notify cabin occupants to jettison passenger door panel and emergency hatch.
7. Hover helicopter until all personnel, including copilot, and survival equipment have been dropped into the water.
8. Proceed downwind approximately 50 yards for ditching of helicopter (pilot only).
9. Ditching should be accomplished from a hover heading into the wind. The helicopter should be lowered vertically into the water, as slowly as possible, and in a level attitude.
10. Immediately upon contact with the water, pull fire emergency handles and apply rotor brake gradually.
11. When the main rotor blades have stopped turning, abandon the helicopter. The pilot can leave through either sliding window in the pilot's compartment.
12. Crew should not inflate raft until all personnel including pilot are at raft. Raft will drift swiftly when inflated.

IMMEDIATE DITCHING.

If for any reason, further flight is not advisable or possible, proceed as instructed in the appropriate emergency procedures in this section. However, contact with the water should, in every instance, be made with as little forward speed as possible. In addition to regular emergency procedures, comply with the following during the approach and flare.

IMMEDIATE DITCHING PROCEDURE.

1. Warn cabin occupants over interphone and with six short rings on the alarm bell.

Note

If the helicopter is equipped with troop seats, notify cabin occupants to check that their safety belts are fastened and to sit with their arms braced against their knees until contact with the water has been made. If the helicopter is not equipped with troop seats, cabin occupants should assume positions aft of any cargo that might be present. If there is no cargo, cabin occupants should brace themselves in any possible manner to prevent or retard any forward movement.

2. Transmit distress message.
3. Shoulder harness inertia reel lock levers —LOCKED.
4. Life vests — Check security.
5. During an autorotative landing, accomplish the following before the flare:
   a. Notify cabin occupants over interphone, and by one long sustained ring of the alarm bell, to brace for ditching.
   b. Fuel shut-off switches — OFF.
   c. Ignition switches — OFF.
   d. Generator switches — OFF.
6. Immediately upon contact with water, apply rotor brake gradually.
7. During a power-on landing, immediately upon contact with the water, pull fire emergency handles, battery switch — OFF, and apply rotor brake lever gradually.
8. Release sliding windows in pilot's compartment and jettison passenger door panel and emergency hatch as briefed.
9. When the main rotor blades have stopped turning, abandon the helicopter. The pilot and copilot leave through either window in the pilot's compartment, and cabin occupants leave through the passenger door or the cabin emergency hatch.

BAILOUT.

If personnel are carried in the cabin, appoint one person to be jumpmaster. He should monitor the interphone to receive the bailout orders, notify the pilot when cabin occupants are prepared to jump, direct the orderly abandonment of the cabin and should be the last to leave the cabin, notifying the pilot just before he jumps.

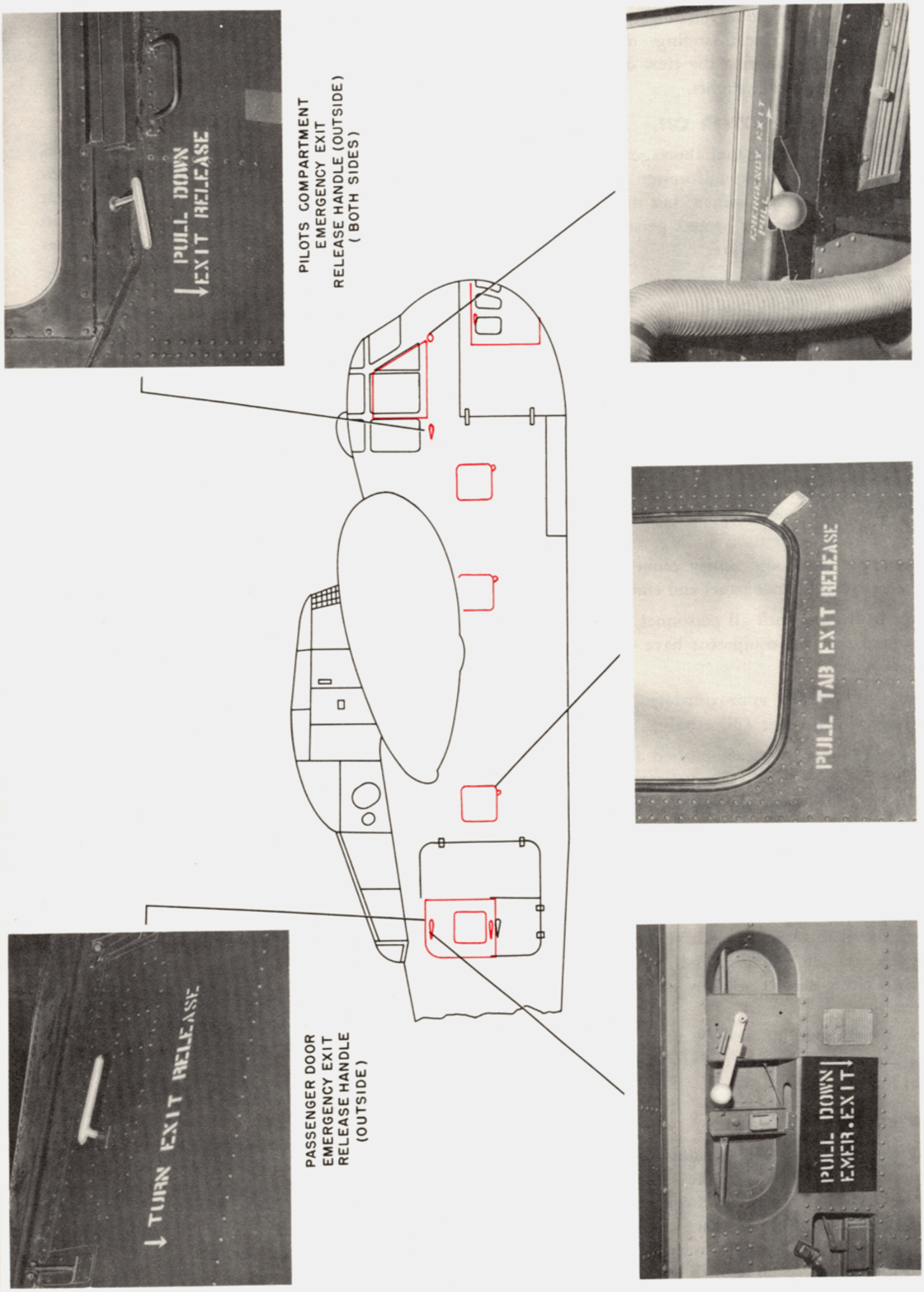


Figure No. 3–2. Emergency Exits and Entrances (Sheet 1)

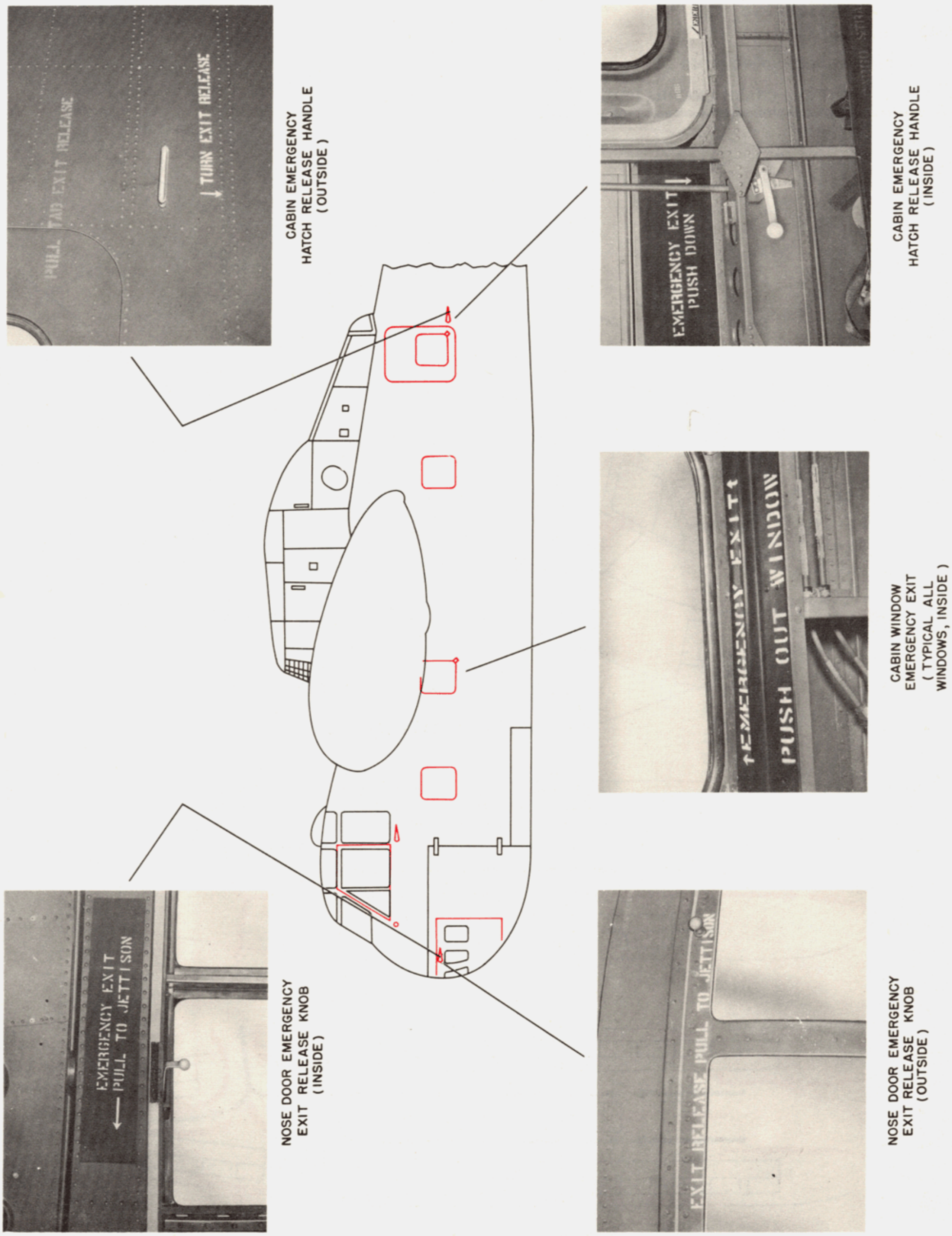


Figure No. 3–2. Emergency Exits and Entrances (Sheet 2)

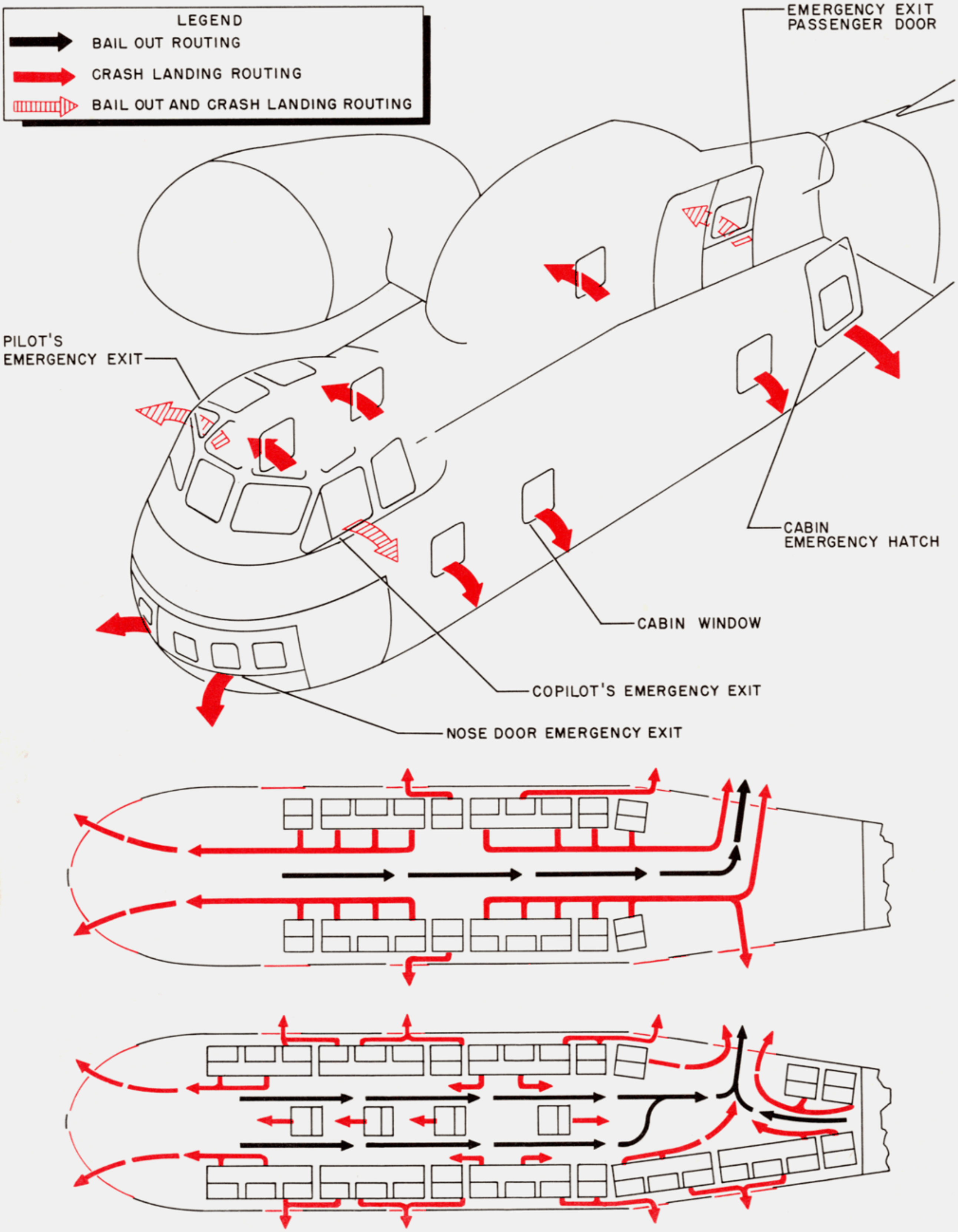
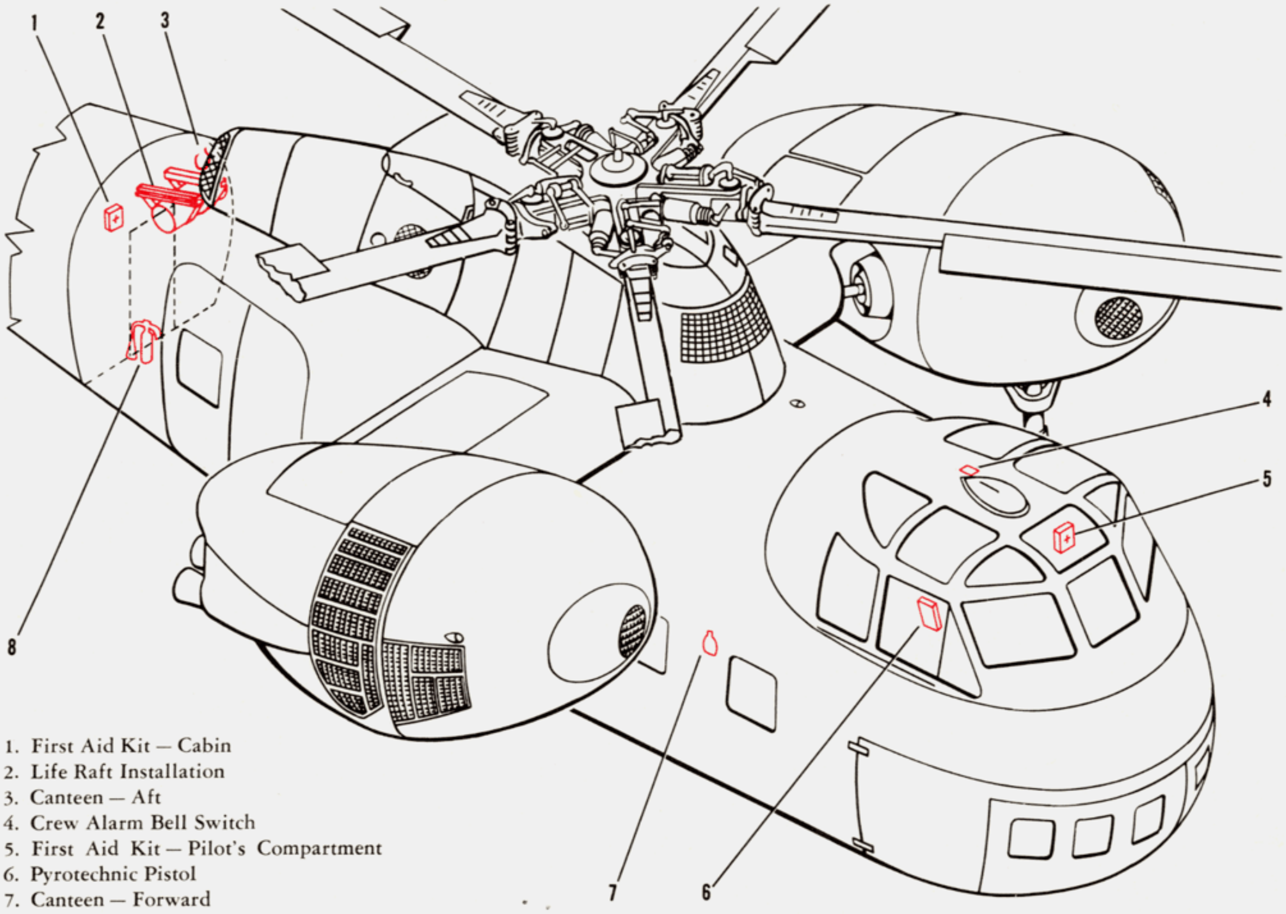


Figure 3—3. Emergency Escape Routing



1. First Aid Kit — Cabin
2. Life Raft Installation
3. Canteen — Aft
4. Crew Alarm Bell Switch
5. First Aid Kit — Pilot's Compartment
6. Pyrotechnic Pistol
7. Canteen — Forward
8. Portable Fire Extinguisher

Figure 3–4. Emergency Equipment

BAILOUT PROCEDURE.

1. Attain a 60 to 70 knot airspeed. Adjust stick trim system to hold helicopter level.
2. Alert cabin occupants over interphone or with the alarm bell.
3. Jettison the sliding windows in the pilot's compartment and the cabin aft passenger door.
4. Receive confirmation that cabin occupants are prepared to jump.
5. Give bailout order over interphone or with alarm bell.
6. Cabin occupants dive out of passenger door head first, feet together, arms close to body, head down. Wait until clear of the helicopter before pulling rip cord in order to avoid fouling the parachute.
7. Pilot and copilot will bail out of their respective windows in pilot's compartment.
   a. Each pilot will crouch in the cockpit.
   b. The pilot will place his left foot on the seat and the copilot will place his right foot on the seat.
   c. Each pilot will place the heel of his other foot on the respective window ledge.
   d. The copilot will kick away from the helicopter without need of sitting on the collective pitch control.
   e. The pilot will transfer the cyclic control stick to his left hand before kicking away from the helicopter.

WARNING

If shoulder straps are entangled in the parachute harness, each pilot will get a definite indication of this condition before getting halfway out of the helicopter.

MAIN GEAR BOX OIL SYSTEM FAILURE.

If the main gear box low pressure warning light should go on, or if main gear box oil temperature exceeds maximum red line temperature in flight, an immediate landing should be made, as the main gear box is not receiving proper lubrication. Flight should not be resumed until the cause has been determined and corrected.

FUEL SYSTEM FAILURE.

FUEL BOOSTER PUMP FAILURE.

If a main tank fuel booster pump should fail, gravity feed and the engine-driven fuel pump should supply fuel under sufficient pressure for normal engine operation. If the fuel pressure is low but not below the normal (green arc) range on the fuel pressure gage, the fuel booster pump switch may be placed in the EMER position. If fuel pressure does not rise, shut the pump off as the low pressure may be caused by a fuel leak. If there should not be sufficient fuel pressure to operate the engine with a failed booster pump, the fuel cross-feed system may be used.

WARNING

Do not place the fuel shut-off switch, for the system having the inoperative pump, in the OFF position. The valve it controls is downstream of the cross-feed valve and will shut off the fuel supply to its engine.

USE OF CROSS-FEED SYSTEM.

Each engine has a separate fuel system which consists essentially of a tank, an electrically operated fuel sump booster pump, and an engine-driven pump. The only connection between the two systems is the cross-feed system. The cross-feed system enables the pilot to operate one engine on fuel from both tanks during single engine operation, or to operate both engines from one tank in event of damage to or loss of fuel from the other tank.

To operate one engine from both tanks:

1. Fuel shut-off switch (for inoperative engine) — OFF.
2. Fuel cross-feed switch — OPEN.
3. Fuel booster pump switches — NORM, or EMER if necessary to maintain fuel pressure.
4. Check for a difference in fuel pressure when operating on each tank by operating first on one fuel pump only, then on the other fuel pump only. If a noticeable difference in pressure exists, it is possible that fuel will flow only from the tank with the higher pressured pump. Check the fuel quantity gages periodically. It may be more satisfactory to operate from one tank at a time.

To operate both engines from one tank:

1. Fuel booster pump switch (for damaged tank) — OFF.
2. Fuel cross-feed switch — OPEN.
3. Fuel booster pump switch (for good tank) — NORM, or EMER if necessary to maintain fuel pressure.

AUXILIARY FUEL SYSTEM FAILURE.

When the warning light, NO TRANSFER, illuminates, it indicates that one of four conditions exists:

1. Normal completion of fuel transfer.
2. Failure of the auxiliary fuel tank pump.
3. Malfunction of the level control valve.
4. Malfunction of the float switch.

In the first case, an accurate account of the elapsed time since fuel transfer was initiated should be sufficient to determine if fuel exhaustion is the cause of the light illuminating. To distinguish between pump failure and malfunction of the level control valve, place the respective auxiliary fuel tank pump switch in the OFF position and allow approximately 5 minutes to elapse. At the end of the period of time, place the pump switch in the ON position. Continued lighting of the no transfer light indicates auxiliary fuel tank pump failure. The fuel remaining will be unavailable. If the no transfer warning light does not remain illuminated when the auxiliary fuel tank pump switch is placed at the ON position, the probable cause of the warning light was a temporary malfunction of the level control valve due to presence of foreign matter and that the float switch actuated the warning light when it turned off the pump. The procedure of shutting off the auxiliary fuel tank pump for approximately 5 minutes is designed to allow the fuel level to fall below the level control valve and in effect, "flush out" the valve. This should dislodge and remove any foreign matter and proper operation of the level control valve should ensue. If the malfunction is not corrected, the "flushing out" should be repeated several times. Should all attempts fail, control of fuel transfer may be accomplished by allowing fuel transfer to continue to a point short of filling the main tanks at which time the auxiliary fuel tank pump switch should be placed at OFF. Transfer may again be commenced when the level of the main tank has lowered appreciably. If the float switch should malfunction and turn off the auxiliary fuel pump, the no transfer warning light will illuminate. Transfer of fuel may be resumed by placing the auxiliary fuel pump switch in the OVERRIDE position. If the OVERRIDE position is used to transfer fuel from an auxiliary fuel tank to a main tank, monitor the fuel quantity gage to preclude overflowing the main tank. In the event of a malfunctioning float switch and level control valve, fuel transfer may be controlled by periodic use of the OVERRIDE and OFF positions of the auxiliary fuel pump switch.

ELECTRICAL POWER SUPPLY SYSTEM FAILURE.

FAILURE OF ONE GENERATOR.

The electrical system is designed to operate on one generator; however, if one generator or one engine should fail, place the generator switch of the inoperative generator in the OFF position and turn off all nonessential electrical equipment.

FAILURE OF BOTH GENERATORS.

If both generators fail, the primary bus will receive power from the battery. Relays will disconnect the secondary bus and the number 2 inverter from the power supply system. If it is necessary to use equipment powered by the secondary bus as well as the primary bus during flight, the battery switch placed in the BATT. EMER position will connect the secondary bus to the battery circuit. Turn off all nonessential electrical equipment. The secondary bus will also receive battery power when the landing gear lever is placed in the DN position.

NUMBER 1 INVERTER FAILURE.

Failure of the number 1 inverter is indicated by the lighting of the instrument power failure warning light and loss of the ac operated flight instruments. The instruments may be restored to operation by placing the instrument power switch in the NO. 2 INV. position and the warning light will go out. With the instrument power switch in the NO. 2 INV. position, all ac equipment will be in operation.

NUMBER 2 INVERTER FAILURE.

The automatic stabilization equipment, IFF equipment, the radar height indicator, and the UHF navigation set will be inoperative if the number 2 inverter fails. The gyro-magnetic compass will be automatically switched to the number 1 inverter if the number 2 inverter fails. The number 2 inverter will be inoperative if both generators fail. Failure of the number 2 inverter is not indicated by the instrument power failure warning light unless the instrument power switch is in the NO. 2 INV. position.

FAILURE OF BOTH INVERTERS.

If both inverters fail, the following instruments will be inoperative:

Fuel Quantity Gages
Manifold Pressure Gages
Fuel Pressure Gages
Engine Oil Pressure Gages
Transmission Oil Pressure Gage
Hydraulic Pressure Gages
Gyro-Magnetic Compass
Attitude Indicators
Automatic Stabilization Equipment
AN/ARA-25 Direction Finder
AN/APN-22 Radar Height Indicator
AN/APX-6 IFF Equipment
AN/ARN-21 UHF Navigation Set

TAIL ROTOR FAILURE WHILE HOVERING (5 TO 10 FEET).

If a tail rotor failure should occur during a hover, simultaneously close the throttles with the twist-grip throttle and raise the collective pitch control to maximum pitch. Lower the collective pitch control smoothly upon ground contact and accomplish normal engine shutdown procedure.

TAIL ROTOR FAILURE IN FLIGHT.

If a tail rotor failure should occur during flight, an attempt should be made to obtain a combination of power and airspeed which will allow the helicopter to maintain a heading. This airspeed will probably be in the vicinity of 70 to 80 knots IAS. A landing site should then be selected and an autorotative approach initiated. The copilot should monitor the control quadrant throttles when the collective pitch is increased to cushion the landing to insure that engine power is not applied at this time. Lower the collective pitch control smoothly upon ground contact and accomplish normal engine shutdown procedures.

UTILITY HYDRAULIC SYSTEM FAILURE.

HYDRAULIC PRESSURE FAILURE.

The utility hydraulic system receives pressure from a pump driven by the left engine. In event of failure of this hydraulic pump, the left engine, or hydraulic pressure in the system, the following equipment will be inoperative:

Landing Gear
Ramp
Nose Doors
Stabilizer
Blade Folding
Tail Rotor Control Pedal Damper
Twist-Grip Throttle Servo
Auxiliary Servo Unit
Wheel Brakes Hydraulic Power Boost

The landing gear can be lowered by means of the emergency hydraulic system (figure 1—41) as covered in **LANDING GEAR FAILURE** in this section. The ramp can be raised manually and lowered by actuating the emergency lock levers. The nose doors can be operated and closed manually. Refer to **NOSE DOORS AND RAMP,** Section IV. The stabilizer will assume the hovering position. Care should be taken in using the tail rotor

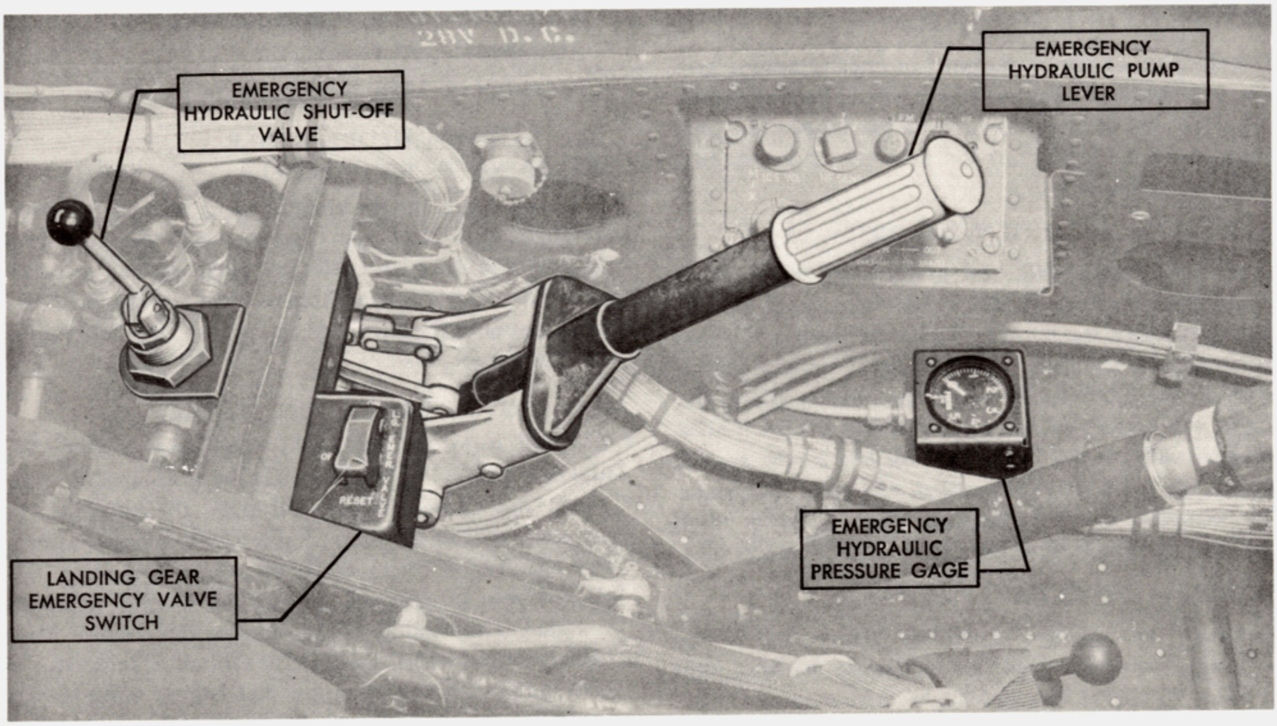


Figure 3—5. Emergency Landing Gear Controls

pedals to avoid abrupt changes in tail rotor pitch as the pedal damper is inoperative. Greater force will be required to operate the twist-grip throttle as the throttle servo is inoperative. The rotor brake may be inoperative if there is excessive loss of fluid from the utility hydraulic system. Although the hydraulic power boost will be unavailable to the wheel brakes, they will be operative to the extent that the pilot can exert foot pressure on the brake pedals.

ELECTRICAL FAILURE OF VALVES.

In the event of electrical power failure, the hydraulic valves of equipment operating from the utility system cannot be actuated electrically. The landing gear can be lowered by use of the emergency hydraulic system. The following equipment can be actuated by positioning the manual override controls:

Ramp

Stabilizer

Nose Doors

Blades Folding

STABILIZER FAILURE.

Stabilizer failure is not critical except possibly at high forward speeds with high gross weight at an extreme cg location. Under these conditions, vibration due to tail buffeting may be encountered as the fuselage assumes an extreme pitch attitude. Reduce speed until the vibration stops, make control movements slowly and smoothly. When landing, hover as little as possible if the stabilizer has failed into the forward flight position.

STABILIZER MALFUNCTION CAUSED BY HYDRAULIC FAILURE.

If pressure should fail in the utility hydraulic system, or if the left engine which drives the utility system hydraulic pump should fail, the stabilizer will be inoperative and will assume the free-swinging hovering position when airspeed is reduced.

STABILIZER MALFUNCTION CAUSED BY ELECTRICAL FAILURE.

In the event of electrical failure, the stabilizer will remain at or return to the forward flight position. Either of the following procedures may be used when it is necessary to place the stabilizer in the hover position:

(1) In the event the secondary bus was de-energized by failure of a power source, restore electrical power to the secondary bus (**ELECTRICAL POWER SUPPLY SYSTEM FAILURE** in this section) and proceed normally or

(2) Push in on the manual override lever on the stabilizer hydraulic valve (figure 1—38). The lever must be held in so long as it is desired to have the stabilizer in the hover position.

If, at the time of the electrical failure, the stabilizer is at its extreme "overcenter" position (leading edge pointing up and aft), the stabilizer override lever on the hydraulic valve is manually positioned to allow the stabilizer to swing free.

PEDAL DAMPER FAILURE.

If the pedal damper should become clogged, the tail rotor control pedals will be restricted in their motion. To free the pedals, place the automatic stabilization servo shut-off switch in the OFF position. If hard left pedal was being applied at the time of restriction, left pedal force must be relieved in order to clear the damper. Release pressure on the left pedal momentarily. After the damper has been cleared, the automatic stabilization servo shut-off switch may be turned on again at the discretion of the pilot. Watch for a reoccurrence of pedal restriction and turn the switch off again if necessary. With the pedal damper turned off, operate the tail rotor control pedals slowly and smoothly to avoid sudden changes in tail rotor thrust. When the pedal damper is turned off, the twist-grip throttle servo and the auxiliary servo unit are inoperative. Greater force will be required to operate the twist-grip throttle.

FLIGHT CONTROL SERVO FAILURE.

Control of the helicopter can be maintained through either the first stage or the second stage flight control hydraulic servo system if one or the other should fail, though prolonged operation on one servo system is not recommended. Land as soon as practicable because control is impossible with both servo systems inoperative. In event of failure of the right engine, the second stage servo system will be inoperative as the second stage servo hydraulic pump is driven by the right engine.

SERVO HYDRAULIC PRESSURE FAILURE.

Loss of hydraulic pressure in either the first or second stage servo systems will be indicated by the lighting of either of the servo hydraulic low pressure warning lights, and by an indication below the normal operating range on the corresponding servo hydraulic pressure gage. When this condition occurs, position the first and second stage servo shut-off switch to turn off the defective system, reduce airspeed to between 60 and 70 knots, and land as soon as practicable. Should the remaining servo system fail, the vibratory loads imposed on the control rods would render the aircraft uncontrollable.

SERVO UNIT MALFUNCTION.

Malfunction of the main or tail rotor servo units during flight will result in erratic behavior of the helicopter, roughness, uncontrollable maneuvers, or locking of the cyclic stick. It is sometimes difficult to determine that the servo units are responsible; or if so, whether the first or second stage is causing the trouble. Both systems may be operating at normal hydraulic pressure and the system that is operating normally will tend to mask the effects of the malfuctioning system. Whenever any of the signs of malfunction are experienced, reduce air speed to 60 to 70 knots to lessen the effects of the malfunction. Turn off first one servo system then the other, by positioning the first and second stage servo shut-off switch, and continue flight on the system which is operating normally.

WARNING

Abrupt maneuvers of the helicopter may develop when the correctly operating system is turned off. When the first and second stage servo shut-off switch is actuated, preparation should be made to return it to the center ON position instantly. Control of the helicopter will very quickly be lost when operating under only one malfunctioning servo system.

Note

Because of the pressure switch interlock, it is impossible to turn off either servo system even though one is malfunctioning, if the hydraulic pressure is below 1500 psi in the remaining system.

LANDING GEAR FAILURE.

In the event of failure of the left engine with consequent loss of the utility hydraulic system, utility hydraulic system failure, or failure of the electrical circuit to the landing gear actuating valve, the landing gear cannot be retracted; however, it can be lowered by use of the emergency landing gear controls (figure 3–5).
To lower the landing gear proceed as follows:

CAUTION

Do not exceed 1500 psi while actuating the emergency pump.

LANDING GEAR — ELECTRICAL MALFUNCTION

MALFUNCTION	POSSIBLE CAUSE	ACTION
I. One indicator shows continuous down and locked (no "barber pole") during retraction or extension. Erratic, up and locked, or down and locked while retracted. Warning light does not come on in handle.	One down and locked limit switch stuck in down and locked position.	1. Close emergency hydraulic valve. 2. Actuate emergency hydraulic pump lever to obtain 1500 psi. 3. Land maintaining 1500 psi. 4. Check down-locks for actual down and locked condition.
II. Both indicators show continuous down and locked (no ."barber pole") during retraction or extension. Erratic indications while retracted. Light does not come on in handle.	Both down and locked limit switches stuck in down and locked positions.	Same as above.
III. One indicator ("barber poled") warning light on in gear handle. Gear apparently fully extended.	One down and locked limit switch failed or out of adjustment.	Pull landing gear control circuit breaker and proceed as above.

Figure 3–6. Landing Gear — Electrical Malfunction

1. Landing gear actuating lever — DN.
2. Emergency hydraulic shut-off valve — CLOSE.
3. Emergency hydraulic pump lever — Extend lever and actuate.

Note

If the gear does not fall free prior to attaining maximum allowable pressure with the emergency hydraulic pump, a sticking actuating valve may be suspected. Place the landing gear emergency valve switch to the ON position. In case of malfunction of the up-lock mechanism, the landing gear may be lowered by having a crew member pull the appropriate up-lock emergency release handle located on the left and right side of the inside cabin walls just aft of the rear wing beam.

CAUTION

If the emergency up-lock release is used to lower the landing gear, the up-locks must be reset before the gear is retracted again.

4. After the gear drops, continue to actuate the emergency pump lever until maximum allowable pressure is attained. Approximately 170 strokes of the pump lever (which can be accomplished in about 2 minutes) are required to lower the gear and seat the down-locks.
5. Execute a landing maintaining full pressure.
6. After landing, check both down lockpins by hand to ascertain that they are firmly extended before opening the emergency hydraulic shut-off valve.

WARNING

If the landing gear emergency valve switch has been utilized, keep the emergency hydraulic shut-off valve closed, the emergency switch ON, and maintain electrical power until the left-hand engine is cut and utility hydraulic pressure decays.

7. Alert the crew chief that the helicopter must be put on jacks until the malfunction is located and corrected.

LANDING GEAR — ELECTRICAL MALFUNCTIONS.

In general, any departure from normal in the operation of the landing gear or the indications in the landing gear control panel (figure 3–6) is sufficient reason to use the emergency hydraulic system. Any indicator not operating ("sticky"), or not going to the mid position ("barber pole"), during the retraction or extension stroke, are typical possibilities. Even though the gear may respond to the placing of the landing gear handle in the DN position, and the indicators show down and locked, any abnormality will warrant the use of the emergency hydraulic system and a landing accomplished while full pressure is maintained. If the landing gear handle warning light remains on, the landing gear control circuit breaker must be pulled before employing the emergency hydraulic system. In all cases, the down-locks are to be checked for proper seating prior to opening the emergency hydraulic valve.

LANDING WITH WHEELS RETRACTED OR IMPROPERLY LOWERED.

To make a successful landing with improperly lowered landing gear is not as difficult with a helicopter as with a conventional aircraft. By proper selection of a landing site and careful hovering and letdown, it is possible to land with minimum danger to personnel or damage to the aircraft. If all attempts to lower the landing gear by actuation of the emergency hydraulic system are unsuccessful, it may be possible to jar the gear loose by an abrupt increase in collective pitch after a shallow dive.

LANDING WITH BOTH WHEELS RETRACTED OR WITH DOWN-LOCKS NOT SEATED.

If a landing must be made with both wheels retracted, or with one or both wheels down but not locked, choose a level spot with no obstructions and preferably with a soft surface such as sand, grass, or bushes.

1. Hover at high power and let down slowly and smoothly with no forward or sideward motion.
2. As soon as the wheels (or wheel) touch, they will be pushed backward as in partial retraction. The helicopter will settle on bottom structure.
3. Reduce rotor speed slowly to note which way, if any, the helicopter will tilt. The greatest damage with a nonwheel landing will occur to the rotor head and blades if the helicopter tips to one side and the blades strike the ground. The blades will touch the ground even if the helicopter rests on the bottom and one nacelle.
4. Maintain control as long as possible with the cyclic control stick as the rotors slow down.
5. Apply the rotor brake gradually when control is no longer effective and the helicopter starts to tip.

LANDING WITH ONE WHEEL RETRACTED OR ONE DOWN-LOCK NOT SEATED.

If only one wheel is down and locked and the other wheel fails to lower or to lock down, even when the emergency hydraulic system is actuated, attempt to raise or unlock the wheel that is down by utility system hydraulic power in order to provide a symmetrical configuration for a safer landing.

1. Utility hydraulic pressure gage — Check for pressure within operating range.
2. Emergency hydraulic shut-off valve — OPEN.
3. Landing gear emergency valve switch (if used) — OFF.
4. Landing gear actuating lever — UP.
5. If the down-lock will not release or the utility hydraulic system is inoperative, head the helicopter into the wind and attempt to touchdown as gently as possible on the one wheel and the opposite side of the fuselage bottom structure.
6. If there is a probability that the helicopter will tip over upon landing, or that the impact will be severe, secure or jettison all loose equipment and hover near the ground to evacuate all cabin personnel. In leaving the helicopter, they should watch out for the main and tail rotor blades, especially under windy conditions.
7. Crew members remaining in the helicopter — Lock shoulder harness inertia reels.

AUTOMATIC STABILIZATION EQUIPMENT FAILURE.

Note

In event of No. 1 engine failure, the automatic stabilization equipment and the auxiliary servo unit will become inoperative as the utility servo hydraulic pump is driven by the No. 1 engine.

POWER SUPPLY FAILURE.

The automatic stabilization equipment will become inoperative in event of failure of the No. 1 engine, both generators, the No. 2 inverter, or the auxiliary servo unit.

MALFUNCTION.

In the event that automatic stabilization signals cause the helicopter to oscillate in pitch, roll, or yaw, or that the barometric altitude control cannot be turned off by pulling the appropriate button, the system should be disengaged by depressing the button, marked AUTO STAB RELEASE, on either cyclic control stick grip. Since the authority of the automatic stabilization equipment on all channels is limited to a fraction of total control travel, any emergency override in case of malfunction may normally be achieved by introducing a control correction. However, if the malfunction is not corrected by depressing the automatic stabilization release switch, the equipment may be eliminated from the flight control system entirely by turning off the automatic stabilization servo shut-off switch. With the automatic stabilization servo shut-off switch unit turned off, care must be used in operating the tail rotor control pedals to avoid sudden changes of tail rotor thrust since the pedal damper is then inoperative. The throttle servo is also inoperative and throttle friction must be overcome manually. Cyclic control stick forces will be greater because of a drag from the auxiliary servo unit. The possibility of a failure, which would not be entirely eliminated by depressing the automatic stabilization release switch, is extremely remote. This information is included so that the pilot would be able to recognize such a failure and not feel that the basic control system has been damaged. Protection against such possibility has been built into the equipment. When the automatic stabilization equipment is nulled mechanically, by using the automatic stabilization release switch on either cyclic stick, all motors will be momentarily commanded to center and then electrical power to the motors will be cut off. Pushing the engage button will again supply power. When the engage button on the automatic stabilization equipment control panel is pulled to place the equipment in standby, the motors are commanded to center, but the electrical power is continuously applied. When disengaged on the panel, an intermittent-type failure of the motor control loop could still effect the flight controls in standby. If this should happen, press the automatic stabilization release switch on the cyclic control stick.

SECTION IV
DESCRIPTION & OPERATION OF AUXILIARY EQUIPMENT

HEATING SYSTEM (WINTERIZATION KIT).

A winterization kit (figure 4—1), consisting of a 600,000 BTU heater, may be installed to provide heat for the cabin and pilot's compartment. Preheating of the engine and main rotor head may also be accomplished. The blower and heater are installed in the heater compartment enclosed by the aft position of the main rotor fairing. A fire detector system is also installed in the heater compartment, two registers for cabin heating are installed over the cabin door. Heater fuel is supplied from the left fuel tank through the same fuel line that supplies the auxiliary power unit engine. An APU and heater fuel manual shut-off valve, fuel solenoid valve and fuel pump are located behind a removable panel on the lower inboard side of the left engine nacelle. The APU and heater fuel manual shut-off valve must be in the open position (handle vertical) before the heater or the APU can obtain fuel. The fuel line leads from the left nacelle to the "high" and "low" heater cycling solenoid valves, located in the cabin ceiling underneath the heater compartment, and then to the heater. The "high" cycling valve meters the fuel to the heater in sufficient quantity to produce heat for maximum cabin heating and when actuated by the engine preheat switch to produce heat to preheat the engine and main rotor head. The "low" cycling valve routes the heater fuel through a metering nozzle that allows a smaller quantity of fuel to flow to the heater producing a lower level of cabin heat. After the heater or engine preheat switch has been turned on, the heater will automatically maintain the required temperatures and will be turned off by thermal cycling switches located in the plenum chamber. The heater exhaust extends through the right side of the main rotor fairing. Outside air is blown through the heater compartment, into the plenum chamber, and then through the ducts and registers to heat the cabin. Heated air is also conducted forward to the pilot's compartment registers and to the windshield and nose door defrosting ducts. Flexible ducts may be connected to engine preheat duct connections on the plenum chamber to preheat the engine and main rotor head. A vent fan is located in the heater compartment to purge it of gasoline fumes, giving additional fire protection to the helicopter. The heater compartment vent fan exhausts air out through the right side of the fairing and is automatically in operation whenever the heater is turned on.

VENTILATING FAN SWITCH.

The ventilating fan switch (figure 4—2), located on the pilot's compartment dome light panel, has two marked positions, VENT and ON. When the switch is in the VENT (center) position, the fan is off; when it is in the ON position, the fan is operating. The ventilating fan switch turns on the heater blower to provide ventilation when the heater is not in use. When the ventilating fan is in the ON position, the blower is operated by direct current from the secondary bus, and protected by a circuit breaker, marked VENT, located on the overhead circuit breaker panel.

CABIN AND PILOT'S COMPARTMENT REGISTERS.

Seven registers are installed in the two cabin ducts which extend along the cabin walls near the ceiling; four registers are installed on the right duct and three on the left duct. Two additional registers are located on the outboard bulkhead of the pilot's compartment just above the floor. Ventilation may be controlled by rotating the knob on each register. The knobs are marked OPEN, with an arrow pointing clockwise.

DEFROSTING CONTROL KNOBS.

Ventilating air going to the four defrosting ducts may be regulated or cut off completely by adjusting four

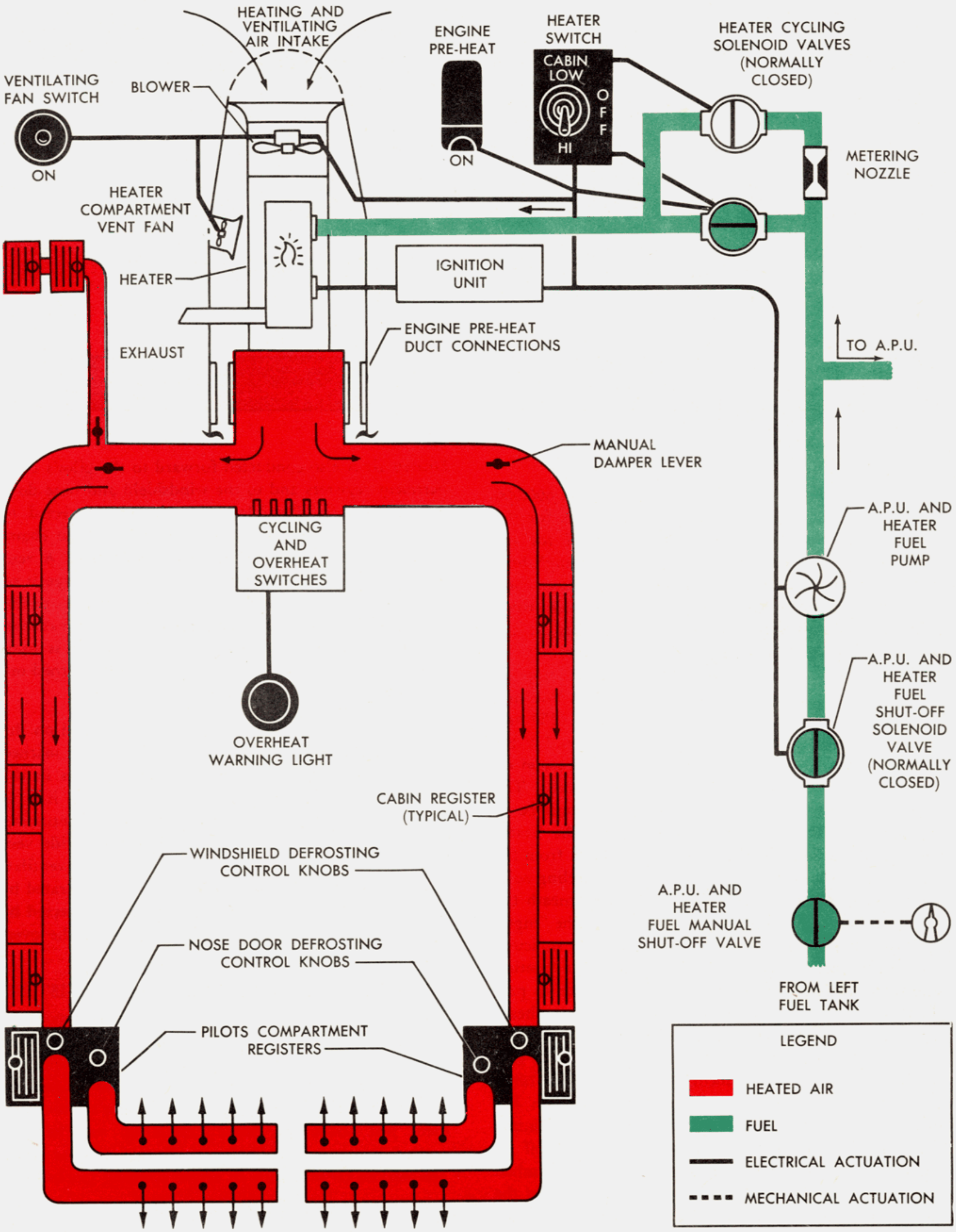


Figure 4—1. Heating System (Winterization Kit)

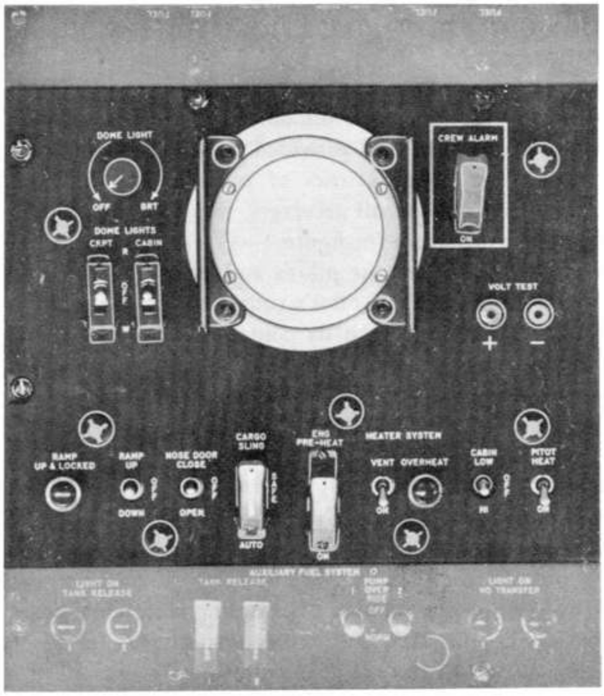


Figure 4—2. Pilot's Compartment Dome Light Panel

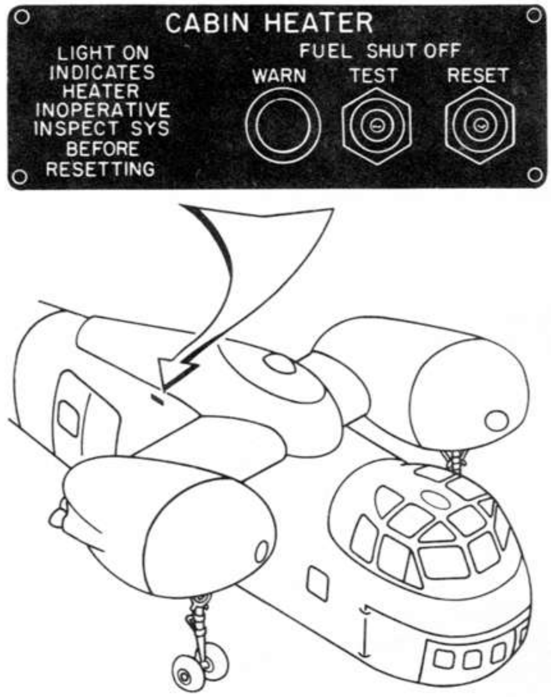


Figure 4—3. Heater Fire Detector Panel

knobs in the pilot's compartment. The upper knobs, located below the left and right pilot's compartment registers, control windshield defrosting, and the lower knobs control nose door defrosting. The spring-loaded knobs are pushed in and rotated clockwise to close off defrosting air, or rotated counterclockwise to increase the flow of air.

CABIN HEATER SWITCH.

Three heater controls, marked HEATER SYSTEM, are grouped together on the pilot's compartment dome light panel. The cabin heater switch (figure 4—2), marked CABIN, has three positions: HI, LOW, and OFF. When the switch is placed in the HI position for heating or defrosting, direct current from the secondary bus operates the heater fuel pump, opens the fuel shut-off valves, turns on the heater compartment vent fan and energizes the heater ignition to produce heat at 140°C (285°F). When the switch is in the low position, the system operates the same as in HI except that heat is produced at only 60°C (140°F). When the heater switch is turned off, both the blower and the heater compartment vent fan continue to run until the temperature in the plenum chamber is reduced to 50°C (120°F), both blower motors are then automatically shut off.

ENGINE PREHEAT SWITCH.

A guarded engine preheat switch (figure 4—2), marked ENG PRE-HEAT, is located on the pilot's compartment dome light panel. When the switch is placed in the ON position for preheating the engine and main rotor head, direct current from the secondary bus operates the heater fuel pump, opens the "high" heater cycling solenoid valve, turns on the heater compartment vent fan, and energizes the heater ignition to produce heat of 177°C (350°F).

OVERHEAT WARNING LIGHT.

A red overheat warning light, marked OVERHEAT, is located on the heater system portion of the pilot's compartment dome light panel. If the heater is operating and all registers are closed so that the heat builds up to 232°C (450°F) in the plenum chamber, or if the temperature rises to this degree for any other reason, an overheat thermal switch on the plenum chamber automatically cuts off the heater ignition and the heater fuel supply and turns on the overheat warning light. When the light goes on, the heater switch or the engine preheat switch should be turned off and the cabin registers opened to allow the blower to disperse the heat. The system should be allowed to cool before using the heater again and enough registers should be opened to disperse the heater output.

HEATER COMPARTMENT FIRE DETECTOR SYSTEM.

A fire detector element is located along the floor of the heater compartment. If a fire should occur in the heater compartment, the fire detector system will shut off the heater fuel supply, blowers, and ignition, and will light a warning light, marked WARN, located on the inboard face of the heater fire detector panel (figure 4–3) on the right side of the cabin, overhead, forward of the cargo door. The panel is marked CABIN HEATER FUEL SHUT-OFF. Two switches, marked TEST and RESET, are also located on the heater fire detector panel. To test the heater fire detector system, hold the test switch on for a few seconds. The warning light should come on, then go off when the test switch is released. When the warning light comes on because of a fire in the heater compartment, the heater is shut off automatically and cannot be operated until the reset switch is actuated.

CAUTION

Do not actuate the heater reset switch during flight. There is no way of inspecting the heating system to ascertain that a fire will not be restarted as soon as the heater is turned on. After landing, make sure that the cause of the heater fire has been found and corrected before actuating the heater reset switches.

NORMAL OPERATION.

1. Heater switch – HI or LO as desired.
2. Register and defrosting knobs – Regulate as necessary.

EMERGENCY OPERATION.

There is no emergency operation of the heating system.

VENTILATING SYSTEM.

Helicopters modified in accordance with Aircraft Service Change No. 10 have the ventilating system removed. However, modification included the installation of a 5-inch louvered vent and cover assembly on each side of the aft cabin. Snap vents have been installed in each cabin window, the cargo door window, and one in each cockpit window. The gun ports can be opened for cabin ventilation, when desired.

PITOT HEATERS.

An electric heater is built into each of the two pitot tubes on the right side of the main gear box fairing to prevent the formation of ice.

PITOT HEATER SWITCH.

A pitot heater switch (figure 4–2), located to the right of the heater control group on the pilot's compartment dome light panel, is marked PITOT HEAT and ON. When this switch is placed in the ON position, direct current from the primary bus operates the pitot heaters. When the switch is returned to the center position, the pitot heaters are turned off. The circuit is protected by a circuit breaker, marked PITOT HEAT, located on the overhead circuit breaker panel.

COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT.

Figure 4–4 lists the communication and navigation radio sets installed and provided for in the helicopter, and the chief characteristics of each. Except for four receiver-transmitters, all receivers, transmitters, amplifiers, and dynamotors (6, figure 1–3) are located on the radio shelves behind the pilot's and copilot's seats, and are accessible from the pilot's compartment. The liaison receiver-transmitter, with its controls, is located at the right of the pilot's seat and the radar height indicator receiver-transmitter is located in the cabin bottom skin structure. The AN/ARC-2 or -2A HF and the AN/ARC-44 transmitter-receivers are located in the aft fuselage section. The location of all antennas is shown in figure 4–6. Twenty-eight volts direct current for all electronic equipment is obtained from the secondary bus. Alternating current for operation of the UHF direction finder is supplied by the number 1 inverter, while alternating current for the IFF, radar height indicator, and the UHF navigation receiver is furnished by the number 2 inverter.

RADIO MASTER SWITCH AND CIRCUIT BREAKER.

All direct current for radio equipment is controlled by a radio master switch, marked RADIO MASTER and ON, located on the overhead switch panel. Power is supplied from the secondary bus through a circuit breaker, marked RADIO MASTER, located on the overhead circuit breaker panel. When the radio master switch is turned on, the interphone system is in operation and the other radio sets may be turned on at the individual control panels.

RADIO CIRCUIT BREAKER PANEL.

From the radio master switch, direct current is fed to the circuit breakers on the radio circuit breaker panel, located at the head of the pilot's compartment ladder on the aft face of the console. A transparent door, hinged at its upper edge, protects the panel from damage and must be swung open to gain access to the circuit breakers. The circuit breakers may be reset by pushing them in, or they may be pulled out to disable completely any radio set.

RADIO FUSE PANEL.

The radio fuses are located on the right side of the control console. The fuses are connected directly to the ac power supply system and are energized whenever the inverters are operating. The fuses are marked as to the radio sets they serve. Six spare fuses comprise the upper row of fuses on the panel.

COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT					
TYPE	DESIGNATION	FUNCTION	PRIMARY OPERATOR	RANGE	LOCATION OF CONTROLS
Interphone	AN/AIC-4A	Intercommunication of crew	Crew members	Interior of helicopter	(1) Pilot — control console (2) Copilot — left side of pilot's compartment (3) Cabin — right side forward of pilot's compartment ladder (4) Cabin — right side forward of cargo door
UHF Transmitter-Receiver	AN/ARC-27A or -55	Short range two-way voice	Pilot or copilot	Line of sight	Control console
HF Transmitter-Receiver	AN/ARC-2 or -2A	Medium range two-way voice	Pilot or copilot	Depends on conditions	Control console
Liaison Set	AN/PRC-8, -9, or -10 (Provisions)	Command liaison	Pilot	Line of sight	At pilot's right
FM Liaison Set	AN/ARC-44	Short range two-way voice	Pilot or copilot	Line of sight to approximately 50 miles	Control console
UHF Direction Finder	AN/ARA-25	Direction finding	Pilot or copilot	Line of sight	Control console UHF control panel — ADF position
UHF Navigation Receiver (DME)	AN/ARN-21	Bearing and distance to beacons	Pilot or copilot	Line of sight	Control console
LF Radio Compass	AN/ARN-41A	Automatic direction finding	Pilot or copilot	Depends upon conditions	Control console
Radar Height Indicator	AN/APN-22	Indicate terrain clearance	Pilot	20,000 feet over water; 10,000 feet over land	Right center of instrument panel
IFF	AN/APX-6B or -6	Identification and tracking	Pilot or copilot	Line of sight	Control console

Figure 4—4. Communication and Associated Electronic Equipment

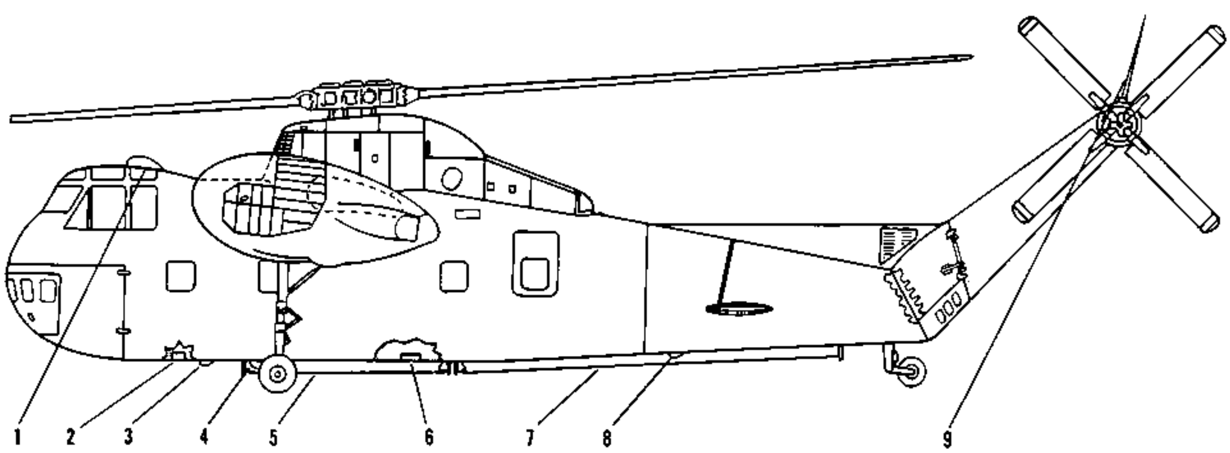
RADIO CONTROL PANELS AND MIXER PANELS. All communication and navigation sets, except the liaison set and the radar height indicator, are controlled from radio control panels (figure 4—8), mounted on the control console located in the center of the pilot's compartment, and are accessible to the pilot and copilot. The description and operation of the controls are described in subsequent paragraphs under headings of the individual radio set. The AN/PRC-8, -9, or -10 liaison set controls are on top of the receiver-transmitter at the pilot's right. The AN/APN-22 radar height indicator is controlled by a knob on the radar height indicator (29, figure 1—12), located on the instrument panel. The UHF set, either the HF or the liaison communication radio set, and the helicopter interphone system are tied together and may be operated simultaneously in various combinations by means of the simplified controls on the pilot's and copilot's mixer panels. The pilot's mixer panel (figure 4—8) is mounted at the right of the control console; the identical copilot's panel is mounted at the copilot's left, underneath the sliding window. Of the three switches, collectively marked RCVR-OUT, the one marked UHF connects the UHF receiver to the headset circuit, and the one marked HF

Figure 4–5 deleted.

connects the HF or the liaison receiver to the headset circuit. The switch, marked FM, connects the FM liaison AN/ARC-44 to the headset circuit. The radio volume control knob adjusts headset signal strength of the three receivers. The ICS volume control knob adjusts headset interphone signal strength. The rotary transmitter selector switch, marked TRANS, connects the microphone circuit to the UHF transmitter when placed in the UHF position and to either the HF or the liaison transmitter when placed in the HF position. The HF sensitivity control knob is inoperative. After the radio master switch is placed in the ON position, and the UHF and either the HF or the liaison receiver-transmitters are turned on and properly tuned from their individual control panels, all communications are controlled from the mixer panels.

MICROPHONE SWITCHES AND HEADSET CONNECTION BOXES.

Two microphone trigger switches, marked RADIO and



1. AN/ARN-41A Direction Finder – Loop
2. AN/ARA-25 UHF Direction Finder
3. AN/APX-6B IFF
4. AN/ARC-27A UHF Transmitter-Receiver (BuNos 140314 through 141617) AN/ARC-55 UHF Transmitter-Receiver (BuNos 145855 and subsequent)
5. AN/ARN-41A Direction Finder – Sense
6. AN/APN-22 Radar Height Indicator
7. AN/ARC-2A HF Transmitter-Receiver
8. AN/ARN-21 DME
9. AN/ARC-44 FM Transmitter-Receiver

Figure 4–6. Antennas (BuNos 140314 and Subsequent)

Figure 4—7 deleted.

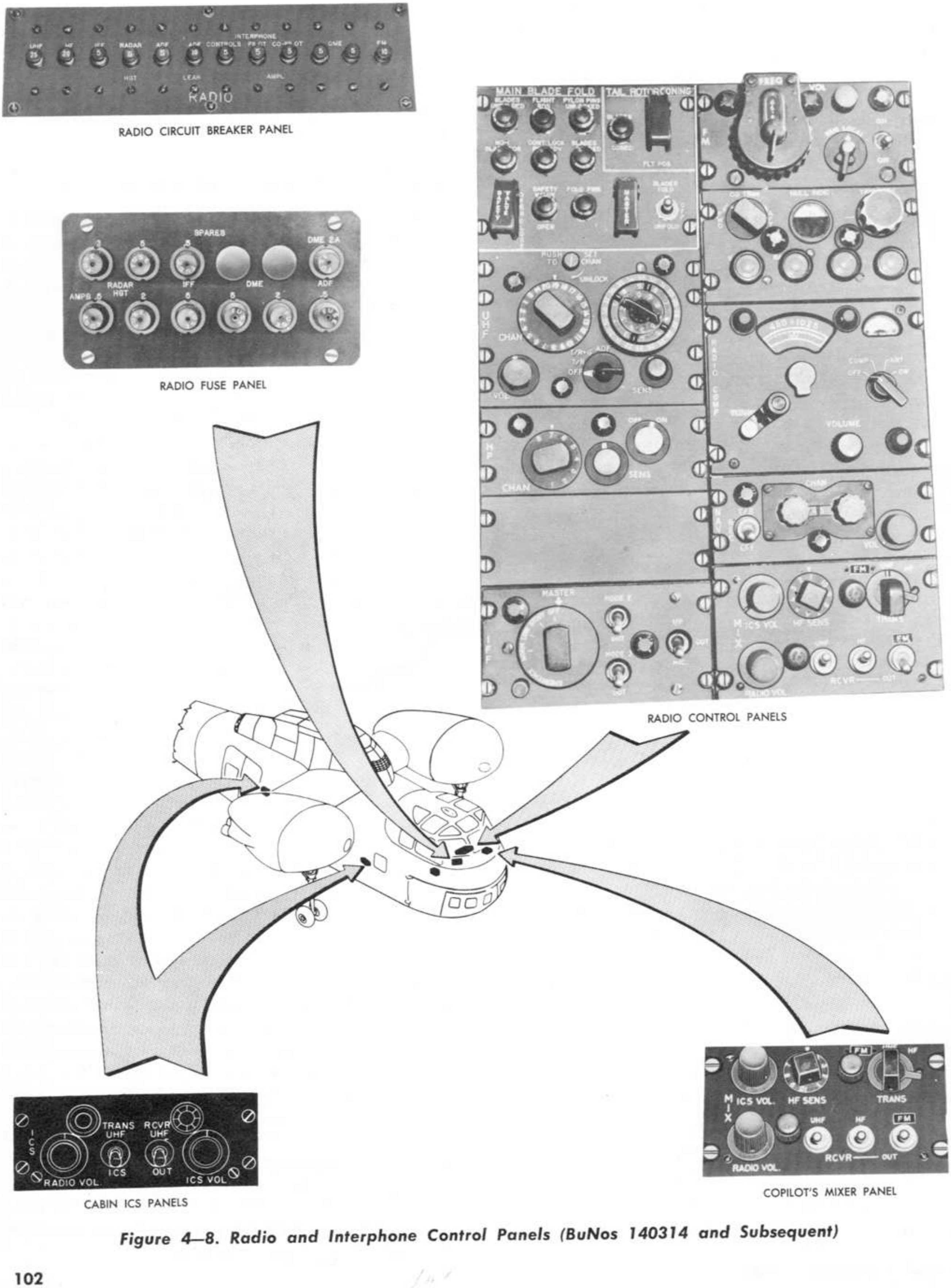


Figure 4—8. Radio and Interphone Control Panels (BuNos 140314 and Subsequent)

ICS, one located on the pilot's and one on the copilot's cyclic control stick grips, connect their respective microphones to the radio transmission circuit when held in the RADIO position, and to the interphone circuit when held in the ICS position. Junction boxes, providing headset and microphone connections as well as hooks for hanging the headsets, are provided overhead in the pilot's compartment for both the pilot and copilot.

CABIN COMMUNICATION STATIONS.

Two communication stations are also provided in the cabin. Both are on the right side of the helicopter, one located just forward of the cargo door, and the other at the forward end of the cabin. These stations are provided with identical ICS control panels, headsets and microphones, a hook for supporting the headsets, and a bracket-mounted portable microphone switch. The controls on these two panels include a transmitter selector switch, which allows choice of transmission over either the UHF radio set or the ICS system, and a receiver-mixer switch, which allows the UHF receiver to be cut in or out at either station. Two volume controls, one for radio and one for interphone, are also provided. Receiver-mixer switches on both the mixer and ICS control panels are collectively marked RCVR, and individually marked with appropriate receiver designations.

INTERPHONE SYSTEM (AN/AIC-4A).

Interphone communications are provided at all four communications stations through the interphone circuit which incorporates two interphone amplifiers. On helicopters modified in accordance with Aircraft Service Change No. 13, interphone amplifiers, AM-40A/AIC, have been replaced with transistorized interphone amplifiers, CW-1003C. Interphone communications are controlled through four control panels (figure 4–8): the pilot's and copilot's mixer panels, and the two ICS control panels, located in the cabin. Interphone reception is automatic at all stations other than for adjustment of the ICS volume controls to provide the desired volume at each station. At either of the two cabin stations, the switch, marked TRANS, must be in the ICS position in order to transmit over interphone. The pilot and copilot then press their control stick microphone switches to the ICS position in order to transmit over interphone.

INTERPHONE OPERATION.

To turn the set on:

1. RADIO MASTER switch – ON.
2. Transmitter switch (cabin stations only) – ICS.
3. ICS volume control – Adjust volume as desired.
4. To transmit from the pilot's or copilot's stations, press the microphone trigger switch on the cyclic control stick grip to the ICS position and speak into the microphone. At either of the two cabin stations, press the portable microphone switch and speak into the microphone.

The interphone set can be turned off by placing the radio master switch in the OFF position.

UHF TRANSMITTER-RECEIVER (AN/ARC-27A).

Radio set, AN/ARC-27A, provides two-way communications between aircraft and ship, aircraft and shore, or between aircraft on any one of 1750 frequency channels in the frequency range of 225.0 to 399.9 megacycles. A guard receiver separate from the main receiver, makes possible the constant monitoring of a guard frequency. The main receiver of the UHF set is used in conjunction with the direction finder, AN/ARA-25. The set is remotely controlled from the radio control panel (figure 4–8), marked UHF, located on the radio control console. Operational controls consist of a four-position function switch, a 22-position channel selector switch, a sensitivity control, and a volume control. The function switch allows for operation as follows: In the OFF position all the equipment is off; in the T/R position the transmitter is on in standby, the main receiver is on, and the automatic direction finder is in standby; in the T/R +G position the transmitter is on in standby, both the main and guard receivers are on, and the automatic direction finder is in standby; in the ADF position the transmitter and the guard receiver are in standby, and both the main receiver and the automatic direction finder are on. The channel selector switch permits the selection of any one of 20 preset main frequency channels for reception and transmission by simply turning the switch to the desired channel. When the selector switch is in the G position, the guard receiver is turned off, but the main receiver and the transmitter are tuned to the guard frequency. Transmission on the guard frequency is possible only when the channel selector switch is in the G position. Reception on the guard frequency is provided through the separate guard receiver, regardless of the setting of the channel selector switch, as long as the function switch is in the TR +G position. When the selector switch is in the M position, the set may be manually tuned to the desired frequency by means of the concentric tuning knobs. The sensitivity control should be set to give the best signal-to-noise ratio. If the sensitivity control is set too high, a "hiss" will be heard in the headsets and incoming signals may not be heard. The volume control should be so adjusted that receiver signal strength is at an understandable and comfortable level with the radio volume control on the mixer panels set to about one-half maximum volume. Receiver volume may then be controlled by the radio volume control on the mixer panels to suit the individual requirements of the pilot and copilot.

UHF TRANSMITTER-RECEIVER OPERATION.

To turn the set on:

1. Radio master switch — ON.
2. UHF receiver mixer switch (mixer panel) — UHF.
3. Function switch — Set for type of operation desired.
4. Channel selector switch — Rotate to desired operating channel.
5. Volume control — Adjust volume after set is warmed up.
6. Sensitivity control — Adjust to best signal-to-noise ratio.
7. Radio volume control (mixer panel) — Adjust to desired signal strength.
8. To transmit — Turn the transmitter selector switch on the mixer panel to UHF, depress the microphone trigger switch on the control stick to the RADIO position, and speak into the microphone.

To turn the set off, turn the function switch to the OFF position.

UHF CHANNEL FREQUENCY CHANGE.

The frequency of any one of the 20 preset channels can be altered by:

1. Channel selector switch — Turn to channel to be changed.
2. Concentric tuning knobs — Rotate to agree with new frequency desired.
3. Channel set knob — Rotate clockwise and push fully in.

The frequency of the channel selected will then correspond to that to which the concentric tuning knobs were placed.

UHF TRANSMITTER-RECEIVER (AN/ARC-55).

Radio set, AN/ARC-55, provides two-way communication between aircraft and ship, aircraft and shore, or between aircraft on any one of 1750 frequency channels in the frequency range of 225.0 to 399.0 megacycles. A guard receiver, separate from the main receiver, makes possible the constant monitoring of a guard frequency. The main receiver of the UHF set is used in conjunction with direction finder group, AN/ARA-25. The set is remotely controlled from the radio control panel (figure 4–8), marked UHF, located on the radio control console. Operational controls consist of a four-position function switch, a 22-position channel selector switch, a sensitivity control, and a volume control. The function switch allows for operation as follows: In the OFF position, all of the equipment is off; in the T/R position the transmitter is on in standby, the main receiver is on, and the automatic direction finder is in standby; in the T/R+G position the transmitter is on in standby, both the main and guard receivers are on, and the automatic direction finder is in standby; in the ADF position the transmitter and the guard receiver are in standby, and the main receiver and the automatic direction finder are on. The channel selector switch permits selection of any one of 20 preset main frequency channels for reception and transmission by simply turning the switch to the desired channel. When the selector switch is in the G position, the guard receiver is turned off, but the main receiver and the transmitter are tuned to the guard frequency. Transmission on the guard frequency is possible only when the channel selector switch is in the G position. Reception on the guard frequency is provided through the separate guard receiver if the function switch is in the TR+G position, regardless of the setting of the channel selector switch. When the channel selector switch is in the M position, the set may be manually tuned to the desired frequency by means of the concentric tuning knobs. The sensitivity control should be set to give the best signal-to-noise ratio. If the sensitivity control is set too high, a "hiss" will be heard in the headsets, and incoming signals may not be heard. The volume control should be so adjusted that receiver signal strength is at an understandable and comfortable level with the radio volume control on the mixer panel set to about one-half maximum volume. Receiver volume may then be controlled by the radio volume control on the mixer panels to suit the individual requirements of the pilot and copilot.

UHF TRANSMITTER-RECEIVER OPERATION.

To turn the set on:

1. Radio master switch — ON.
2. UHF receiver-mixer switch (mixer panel) — UHF.
3. Function switch — Set for type of operation desired.
4. Channel selector switch — Rotate to desired operating channel.
5. Radio volume control (mixer panel) — Adjust to warmed up.
6. Sensitivity control — Adjust to best signal-to-noise ratio.
7. Radio volume control (mixer panel) — Adjust to desired signal strength.
8. To transmit — Turn the transmitter selector switch on the mixer panel to UHF, depress the microphone trigger switch on the control stick to the radio position, and speak into the microphone.

To turn the set off, turn the function switch to the OFF position.

UHF EMERGENCY OPERATION.

If the channel selector switch will not channel to the desired frequency, turn the switch to the M position and tune the set manually with the concentric tuning knobs.

HF TRANSMITTER-RECEIVER (AN/ARC -2 OR -2A).

Radio set, AN/ARC-2A, operates in the frequency range of 2.1 to 9.05 megacycles, and may be used for either voice, CW, or MCW signal reception and transmission. Choice of the type of signal to be used is made by turning the control switch, marked EMISSION, located on the front panel of the receiver-transmitter, to the CW, MCW, or VOICE position as desired. The CW position of this switch provides for CW transmission and either CW or MCW reception; the MCW position provides for MCW transmission and either MCW or voice reception; the VOICE position provides for voice transmission and either voice or MCW reception. The receiver-transmitter (1, figure 1–3) is located on the forward right radio deck. Receiver volume is controlled by the radio volume control on the pilot's and copilot's mixer panels to suit individual requirements. All other operational controls are located on the remote control panel (figure 4–8), marked HF, mounted on the radio control console. These controls consist of an eight-position channel selector switch and an OFF-ON power switch.

HF TRANSMITTER-RECEIVER OPERATION.

To turn the set on:

1. Radio master switch – ON.
2. HF receiver-mixer switch (mixer panel) – HF.
3. Power switch (HF panel) – Rotate clockwise until it catches.
4. Channel selector switch – Rotate to desired operating channel.
5. Radio volume control knob (mixer panel) – Adjust volume as desired after set has warmed up.
6. To transmit – Turn transmitter selector switch on the mixer panel to the HF position, depress the microphone trigger switch on the control stick to the RADIO position and speak into the microphone.

To turn the set off, push the power switch all the way in toward the panel.

Note

Do not attempt to manually rotate the power switch to the OFF position.

FM LIAISON SET (AN/ARC-44).

Radio set, AN/ARC-44, is an FM transmitter-receiver operating in a frequency range of 24.0 to 52.0 megacycles. It provides for two-way voice communication between aircraft or between aircraft and a ground station and is limited by line-of-sight considerations to approximately 50 miles. The set is remotely controlled from the radio control panel (figure 4–8), marked FM, located on the forward right-hand corner of the radio control console, and utilizes a whip antenna, located on top of the tail pylon. The operating controls consist of an ON-OFF switch, a tuning knob with a frequency indicating window, a volume control, and a two-position switch, marked REM and LOCAL, that is inoperative and should be kept in the LOCAL position. Connection of the radio set to the mixer panel is accomplished by placing the selector switches, marked TRANS and RCVR (figure 4–8), in the FM position. The AN/ARC-44 is also equipped with a squelch-disable switch (47, figure 1–12), mounted on the lower center section of the instrument panel. Marked with two positions, ON and OFF, the switch is normally in the ON position. This filters out background and static noise in the headsets, but permits medium and strong signals to be heard more clearly. If it is desired to pick up a weak signal, the receiver is rendered more sensitive, in spite of the increased noise, by turning the squelch-disable switch OFF.

FM LIAISON SET OPERATION.

To turn the set on and receive:

1. Radio master switch (overhead switch panel) – ON.
2. ON-OFF switch (FM panel) – ON.
3. Receiver switch (mixer panel) – FM position.
4. Tuning knob (FM panel) – Turn to desired frequency.
5. Squelch-disable switch (instrument panel) – OFF if greater sensitivity is desired.
6. Volume control (mixer panel) – As desired.

To transmit:

7. Transmitter selector switch (mixer panel) – FM position.
8. Microphone trigger switch (cyclic control stick grip) – Press to RADIO position.

To turn the set off, place the ON-OFF switch in the OFF position.

UHF DIRECTION FINDER GROUP (AN/ARA-25).

Direction finder group, AN/ARA-25, uses signals received by radio set, AN/ARC-27A or AN/ARC-55, within the range of 225 to 400 megacycles to indicate the bearings and to home on radio signal sources. When a signal is being received, a 100-cycle-per-second tone is heard, regardless of whether the incoming signal is amplitude-modulated or unmodulated. The 100-cycle per-second tone is produced by a chopper in the antenna installation. The equipment is controlled from the radio control panel (figure 4—8) of the UHF radio set, AN/ARC-27A or AN/ARC-55, located on the radio control console. When the direction finder is in operation and a signal is being received, the relative bearing of the transmitting station will be indicated by the single-barred pointer, numbered 1, on both pilot's and co-pilot's radio-magnetic indicators, located on the instrument panel. The magnetic bearing of the transmitting station will be indicated on the rotating compass card under the pointer. As the radio compass, AN/ARN-41A, is connected to the double-barred pointer, numbered 2, on the radio-magnetic indicators, simultaneous bearings on two stations may be obtained by using the direction finder group and the radio compass simultaneously.

Note

On BuNos 145865 through 145875, and helicopters modified in accordance with Aircraft Service Change No. 37, the UHF radio direction finder, AN/ARA-25, and the low frequency radio compass, AN/ARN-41A, are connected to the No. 1 pointer on the radio magnetic indicator and either may be selected for use. The AN/ARN-41A is automatically turned off by a relay-operated transfer switch when the AN/ARA-25 is turned on. The UHF radio navigation set, AN/ARN-21, is connected to the No. 2 pointer of the radio magnetic indicator, allowing continuous operation of the AN/ARN-21.

UHF DIRECTION FINDER GROUP OPERATION.
To turn the set on:

1. Radio master switch — ON.
2. UHF receiver-mixer switch (mixer panel) — UHF.
3. Function switch (UHF panel) — ADF.
4. Channel selector switch (UHF panel) — Select desired channel.
5. Volume control — Adjust to give the best bearing indication on the radio magnetic indicator.

To turn the set off, rotate the function switch to the OFF position or to any position other than ADF.

Note

To call a station and request transmission of a homing signal, it is necessary to turn the function switch (UHF panel) to a transmitting position (T/R or T/R+G).

UHF NAVIGATION SET (AN/ARN-21).

Radio navigation set, AN/ARN-21, provides both bearing and distance indications to any selected AN/URN-3 surface navigation beacon located within a line-of-sight distance of up to 195 nautical miles. The set combines the indications of three instruments: the radio magnetic indicator, the course indicator, and the range indicator. The double-barred pointer, numbered 2, of the radio magnetic indicator, (4 and 16, figure 1—12) will give the relative bearing of the selected beacon with the magnetic bearing appearing on the compass card under the pointer. The LF radio compass, AN/ARN-41A, is connected to the same pointer, but the radio compass is automatically turned off by a relay-operated switch when the navigation set, AN/ARN-21, is turned on.

The course indicator (14, figure 1—12) provides a means of setting in a course to the beacon by a knob, marked SET. The selected course appears in the course window. Deviations from this course are indicated by a vertical bar and a relative heading pointer. The ambiguity window will show TO or FROM, indicating whether the selected course is to or from the beacon. A marker beacon light is provided when the set is used in conjunction with ILS. Two warning flags are installed to warn against the use of the cross pointer indications when the set is incorrectly tuned or malfunctioning. The range indicator (15, figure 1—12), marked DISTANCE NAUTICAL (miles), indicates distance to the beacon. When correct distance information cannot be determined, the range indicator will "search." "Searching" is evidenced by rapidly rotating numerals and a red bar extending across them. The set is controlled from a radio control panel (figure 4–8), marked NAV, located on the control console. Controls include a three-position function switch, marked OFF, REC, and T/R; two channel selector knobs, marked CHAN; and a volume control, marked VOL. When the function switch is set in the REC position, only the receiver operates and only bearings are indicated. When the function switch is in the T/R position, both the receiver and the transmitter operate and both bearings and distances are indicated. The transmitter sends out a pulsed interrogation signal which triggers a reply signal from the selected beacon. The navigation set measures the time lapse between the interrogation signal and the reply and converts this into a distance indication. Identification of the selected beacon is accomplished by listening to the call letters spelled out in International Morse Code over the regular headphones. The strength of this tone signal is adjusted by the volume control and may be heard, regardless of the position of the controls on the mixer panel. The left-hand channel selector knob is used to set in the hundreds and tens of the beacon channel number, and the right-hand knob to set in the units. Combinations of dial settings from 00 to 129 may be made. However, the equipment only operates on channels 01 to 126, a total of 126 channels. The transmitter operates in the frequency range of 1025 to 1150 megacycles. The receiver operates in the ranges of 962 to 1024, and 1151 to 1213 megacycles.

Note

On BuNos 145865 through 145875, and helicopters modified in accordance with Aircraft Service Change No. 37, the UHF radio direction finder, AN/ARA-25, and the low frequency radio compass, AN/ARN-41A, are connected to the No. 1 pointer on the radio magnetic indicator and either may be selected for use. The AN/ARN-41A is automatically turned off by a relay-operated transfer switch when the AN/ARA-25 is turned on. The UHF radio navigation set, AN/ARN-21, is connected to the No. 2 pointer of the radio magnetic indicator, allowing continuous operation of the AN/ARN-21.

UHF RADIO NAVIGATION SET OPERATION.

To turn the set on:

1. Radio master switch – ON.
2. Function switch – REC or T/R.
3. Channel selector knobs – Select desired channel.

CAUTION

No attempt should be made, at any time, to set the CHAN dial below channel 01 or above channel 126.

4. Volume control – Adjust as desired.

CAUTION

Always check the channel setting and the call letters to make certain the bearing and range indications are those of the desired beacon, and not of another beacon at a different location.

To turn the set off, turn the function switch to the OFF position.

LF RADIO COMPASS (AN/ARN-41A).

The LF radio compass is a combination automatic direction finder and radio receiver covering the frequency range of 190 to 1725 kilocycles in three bands. A radio control panel (figure 4–8) on the console, marked RADIO COMPASS, contains the following controls: a volume control; a function switch with four positions, OFF, COMP, ANT, and CW; a tuning control; a three-position band switch with a calibrated frequency dial above it and a tuning indicator. When the function switch is in the ANT position, the equipment functions as a standard radio receiver, loop antenna and automatic circuits are inoperative, and the tuning meter is not used. When the function switch is in the COMP position, the equipment functions as an automatic direction finder using both the loop and sense antennas; the tuning knob is adjusted to give a maximum reading on the tuning indicator for any given station, and the relative bearing of the station is automatically indicated by the double-barred pointer, numbered 2, on the pilot's and copilot's radio magnetic indicators (4 and 16, figure 1–12), located on the instrument panel. The magnetic bearing of the transmitting station will be indicated on the rotating compass card under the pointer. When the UHF navigation set, AN/ARN-21, is in operation, it is connected to this same pointer, and the radio compass is automatically turned off by a relay. As the UHF direction finder group, AN/ARA-25, is connected to the single-barred pointer, numbered 1, on the radio magnetic indicators, simultaneous bearings on two stations may be obtained by using the direction finder group and the radio compass simultaneously. The radio compass is connected directly to the headset circuit and may be heard, regardless of the position of the controls on the mixer panel. The function switch may be placed in the CW position to turn on the beat frequency oscillator (BFO) as an aid in tuning a station too weak to hear, but still strong enough for the equipment to perform as a direction finder. When the BFO is used, tune back and forth around the desired station and listen for zero beat, on either side of which a whistle will be heard. When zero beat point is reached, the tuning indicator should simultaneously show maximum indication and the BFO may be switched off. The desired operating band is selected by means of the band switch, and the calibrated frequency dial automatically shows only the frequencies included in the band selected. The radio compass is operative whenever the radio master switch is ON. If the radio compass causes headset interference with other receivers, the volume may be turned down. At night, it is important not to use bearings on stations located more than 100 miles away, as serious inaccuracies may result from doing so.

CAUTION

Do not operate the equipment with the loop motor disconnected.

Note

On BuNos 145865 through 145875, and helicopters modified in accordance with Aircraft Service Change No. 37, the UHF radio direction finder, AN/ARA-25, and the low frequency radio compass, AN/ARN-41A, are connected to the No. 1 pointer on the radio magnetic indicator and either may be selected for use. The AN/ARN-41A is automatically turned off by a relay-operated transfer switch when the AN/ARA-25 is turned on. The UHF radio navigation set, AN/ARN-21, is connected to the No. 2 pointer of the radio magnetic indicator, allowing continuous operation of the AN/ARN-21.

LF RADIO COMPASS OPERATION.

To turn the set on for use as a standard radio receiver:

1. Radio master switch – ON.
2. Function switch – ANT.
3. Band switch – Set to desired operating band.
4. Tuning control – Tune to desired station.
5. Volume control – Adjust as necessary.

To turn the set on for use as an automatic direction finder:

1. Radio master switch – ON.
2. Function switch – COMP.
3. Band switch – Set to desired operating band.
4. Tuning control – Tune to desired station and for maximum reading on the tuning indicator.
5. Function switch – CW (if BFO needed to tune set).
6. Volume control – Adjust as necessary.

The radio compass is turned off by placing the function switch in the OFF position.

SIMULTANEOUS OPERATION.

Communication equipment in the helicopter includes three receiver-transmitters (UHF, HF, and the liaison set) and the interphone system. Interphone reception is automatic at all stations except for volume level which is independently adjustable at each station. When the interphone circuit is energized, incoming radio signals are automatically reduced somewhat in level in order to prevent garbling of interphone communications. Provided that operational controls on the UHF and on either the HF or the liaison control panels are properly set, independent reception and/or transmission through the sets is possible by either the pilot or copilot by use of the simplified radio controls on the mixer panels. However, it is impossible to operate the HF receiver-transmitter or the liaison set independently (without garbled reception) if both are turned on simultaneously because both are connected to the HF receiver switches and the HF position of the transmitter selector switches on the mixer panels.

RADAR HEIGHT INDICATOR (AN/APN-22).

Radar height indicator, AN/APN-22, provides continuous indication of the height (terrain clearance) of the helicopter from 0 to 20000 feet over water, and from 0 to 10000 feet over land. Altitude in feet is indicated by the radar height indicator (29, figure 1–22), mounted on the instrument panel in front of the pilot. The limit control marker on the indicator is continuously adjustable throughout the range of the equipment by means of the limit marker control knob, marked LIMIT, which also serves as a power switch. It is located at the lower left of the indicator. A red warning light, at the lower right of the indicator, shows when the helicopter is at or below the preselected height indicated by the limit control marker. The equipment is turned on and off by means of the power switch which is combined with the limit marker control. This is the only control required for operation of the equipment. A reliability circuit disables the indicator and puts the pointer behind a mask to prevent the pilot from using the indication when the signal is too weak to provide reliable operation. The fuse and the circuit breakers, marked RADAR, located on the radio fuse and circuit breaker panel, protect the height indicator circuits from overload. Fuses to protect the AN/APN-22 primary circuits are located in the junction box at the rear of the mounting for the control amplifier, AM-291/APN-22, which is on the radio deck (6, figure 1–3) behind the pilot. Spare primary power fuses are located on a panel fixed to the underside of the mounting.

RADAR HEIGHT INDICATOR OPERATION.

To turn the set on:

1. Radio master switch – ON.
2. Power switch – ON.
3. Limit marker control knob – Set limit marker to desired altitude.

To turn the set off, turn the power switch to the OFF position.

IFF RADAR IDENTIFICATION SET (AN/APX-6 OR -6B).

The radar identification set, AN/APX-6 or -6B, is an airborne transponder which may be operated to provide an electronic identification and recognition. The purpose of the AN/APX-6 or -6B is:

1. To identify the helicopter in which it is installed as friendly, when correctly challenged by an interrogator-responder associated with friendly shipboard or airborne radar.
2. To permit surface tracking and control of the helicopter in which it is installed.

All controls required for operation of the AN/APX-6 or -6B equipment are located on the radio control panel (figure 4–8), marked IFF, mounted on the control console. The blank panel immediately forward of the IFF control panel on the console is for the installation of equipment to be operated in conjunction with the AN/APX-6 or -6B.

IFF RADAR IDENTIFICATION SET OPERATION.

To turn the set on:

1. Radio master switch – ON.
2. Master selector switch – NORM. (To maintain equipment ready for instant use but inoperative, rotate the master selector switch to STDBY.)
3. To indicate emergency or distress – Depress red dial stop button while turning the master selector switch to the EMERGENCY position.
4. Do not turn the master selector switch to the LOW position except upon proper authorization.

To turn the set off, rotate the master selector switch to the OFF position.

EMERGENCY OPERATION – POWER FAILURE.

In the event of failure of one generator (or failure of one engine rendering a generator inoperative), as indicated by the ammeter pointer of either volt-ammeter fluctuating or returning to zero, all electronic equipment will remain operative. Turn off all nonessential equipment to prevent overloading the remaining generator. In the event of failure of both generators, dc and ac power will be lost. Power may be restored by placing the battery switch in the BATT EMER position. Turn off nonessential equipment in order to prolong the life of the battery. It is also possible to operate the radios on power from the auxiliary power unit. In case of failure of the number 1 inverter, alternating current for operation of the AN/ARA-25 direction finder, the course indicators, and the range indicator will be cut off. When the flight instrument power switch (24, figure 1–12) is placed in the NO. 2 INV. position, this equipment will be restored to operation. In case of failure of the number 2 inverter during flight, the IFF, radar height indicator, and the UHF navigation set (AN/ARN-21) are inoperative and cannot be restored.

LIGHTING EQUIPMENT.

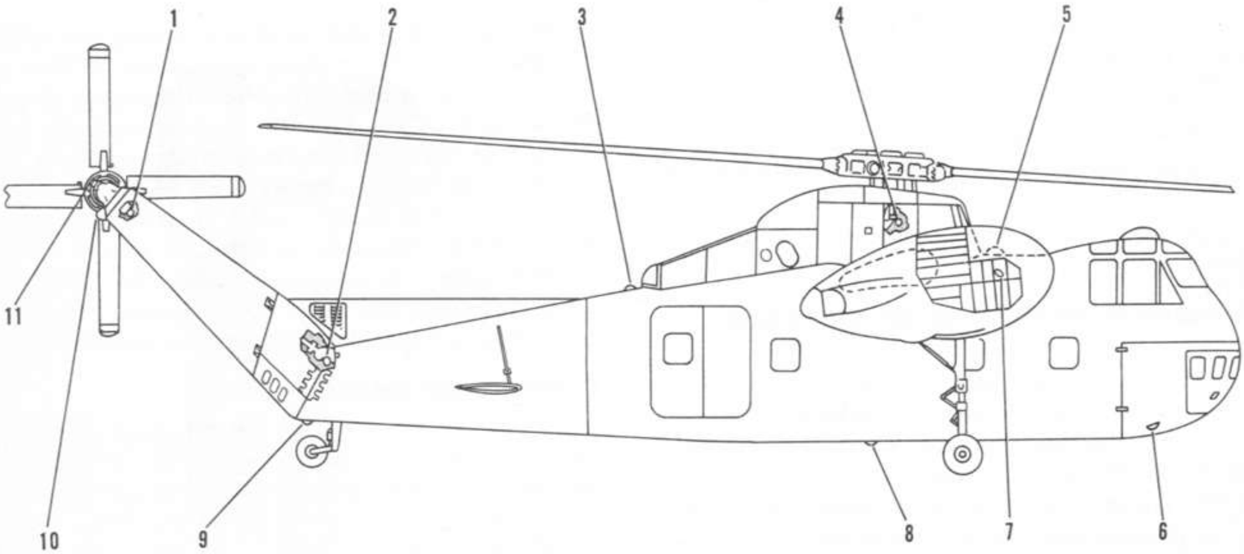
All lights operate on direct current supplied by the primary, secondary, or battery buses. The parking light, gear box inspection lights, and the pilot's compartment dome light operate from the battery bus which is connected directly to the battery. The instrument panel, overhead panel, console, quadrant lights, and the spotlights operate from the primary bus. All other lights operate from the secondary bus. All lights are protected by circuit breakers. The primary and secondary bus circuit breakers are located on the overhead circuit breaker panel (figure 1–24) in the pilot's compartment; the battery bus circuit breakers (figure 1–24) are located in the box inboard of the main dc power junction box.

PARKING AND GEAR BOX OIL LEVEL INSPECTION LIGHT SWITCH.

A parking and gear box oil level inspection light switch (figure 4–10), marked TAIL PARKING LIGHT, OFF, and XMSN OIL LEVEL INSP LTS, is located in a switch box forward of the cabin door on the right side of the cabin. When the switch is placed in the TAIL PARKING LIGHT position, it lights an amber parking light (9, figure 4–9), located under the tail cone aft of the tail wheel. When the switch is placed in the XMSN OIL LEVEL position, it illuminates gear box inspection lights (1, 2, and 4, figure 4–9) at the oil level inspection windows of the main, intermediate, and tail gear boxes, and a red pilot light on the side of the switch box. The switch operates from the battery bus so that these lights may be turned on when the main electrical power is shut off.

POSITION AND FUSELAGE LIGHT SWITCHES.

Two position and fuselage light switches (figure 1–10), marked EXTERIOR LIGHTS, POSITION, located on the overhead switch panel in the pilot's compartment, control the three position lights and the two fuselage lights. The right switch is marked FLASH, OFF, STEADY and the left switch is marked DIM, BRT. A red position light (7, figure 4–9) is located on the outboard side of the left nacelle, a green position light is located on the right nacelle. The third position light is a white light (10, figure 4–9) located at the top of the tail



1. Tail Gear Box Inspection Light
2. Intermediate Gear Box Inspection Light
3. Formation Light
4. Main Gear Box Inspection Light
5. Top Fuselage Light (white)
6. Landing Light
7. Side Position Lights (green on right and red on left)
8. Bottom Fuselage Light (white)
9. Parking Light (amber)
10. Aft Position Light (white)
11. Rotating Anti-Collision Light

Figure 4–9. Exterior Lights and Gear Box Inspection Lights

rotor pylon. The fuselage lights (5 and 8, figure 4--9) are both white, one located on top, forward of the main rotor fairing, the other located on the bottom, aft of the hatchway. The position and fuselage lights operate on direct current from the secondary bus and are protected by a circuit breaker, marked POS and FORM, located on the overhead circuit breaker panel. When the right-hand switch is placed in the FLASH position, the position and fuselage lights flash simultaneously at 80 cycles per minute. When it is placed in the STEADY position the lights burn steady. The position and fuselage lights will burn dimly or brightly in accordance with the position of the left switch, marked DIM, BRT.

Figure 4–10. Parking and Gear Box Inspection Light Switch

ROTATING ANTI-COLLISION LIGHT SWITCH.

On BuNos 145855 and subsequent, and helicopters modified in accordance with Aircraft Service Change No. 21-B, a rotating anti-collision light switch (figure 1–10), marked ROT LT, OFF, and ON, located on the exterior light portion of the overhead switch panel, controls a rotating anti-collision red light (11, figure 4–9), located on top of the pylon. To operate the rotating anti-collision light, the position light switch must be placed in either the FLASH or STEADY position in addition to placing the anti-collision light switch in the ON position. The rotating anti-collision light switch operates on direct current from the secondary bus and is protected by a circuit breaker, marked ROT LIGHT, located on the overhead circuit breaker panel.

Note

The rotating anti-collision light should be turned OFF during flight under actual instrument conditions. With the light on during instrument conditions, the pilot could experience vertigo as a result of the rotating reflections of the light against the clouds. In addition, the light would be ineffective as an anti-collision light during instrument conditions since it could not be observed by pilots of other aircraft.

FORMATION LIGHT SWITCH.

A formation light switch (figure 1—10), marked FORM, located on the overhead switch panel, at the left of the position light switches, controls a formation light (3, figure 4—9), located on top of the fuselage behind the main rotor fairing. The marked positions are DIM, OFF, and BRT.

LANDING LIGHT SWITCHES.

Two landing light switches (figure 4—11), marked LDG LT, located in a box at the forward end of the pilot's collective pitch control, are used to operate a landing light (6, figure 4—9), located in the right nose door. A toggle switch, marked MASTER, OFF, and RETRACT, must be placed in the MASTER position before a spring-loaded four-way thumb switch, marked EXTEND, RETRACT, L, and R, is operative. The toggle switch turns on the lamp and the thumb switch controls direction of the light beam. To extend the landing light from its stowed position to permit it to shine forward, hold the thumb switch in the EXTEND position. To rotate the light to the left or right, hold the switch at L or R; the light will come to rest pointing in the desired direction when thumb pressure is removed from the switch. If it is desired to depress the light beam to shine on a spot closer to the helicopter or directly downward, hold the thumb switch at RETRACT until the desired angle is attained, then release the switch. When it is desired to turn off and stow the landing light after use, it is only necessary to place the toggle master switch in the RETRACT position. The light will go out and will retract and rotate automatically to the stowed position. The landing light is also used as a flood and loading light. The light beam may be pointed in any direction from the horizon to vertically downward through 360 degrees around the helicopter and may be pointed upward approximately 22 degrees in the forward semicircle. The right nacelle and the main rotor blade tips may be illuminated.

PILOT'S COMPARTMENT DOME LIGHT SWITCH AND RHEOSTAT.

A pilot's compartment dome light guarded switch (figure 4—2), marked DOME LIGHTS, CKPT, located at the left rear on the pilot's compartment dome light panel, controls the dome light located in the center of the panel. The switch has three marked positions; R, OFF, and W. The light contains a red and white lamp. To turn on the red lamp, place the switch in the R position; to turn on the white lamp, open the guard and place the switch in the W position. The intensity of the red lamp may be varied by a rheostat, marked DOME LIGHT, OFF, BRT, located at the left of the light.

INSTRUMENT LIGHT RHEOSTATS.

The individual instrument lights are connected in two circuits. One circuit lights the flight instruments and the standby compass and is controlled by a rheostat (figure 1—10), marked FLIGHT INST LIGHTS, OFF, and BRT, located on the overhead switch panel. The standby compass has, in addition, a separate ON, OFF switch, located on the right side of the compass mounting bracket, so that the compass light may be turned off when the other flight instrument lights are on. When the flight instrument lights are turned on, most of the warning lights are automatically dimmed. The second circuit lights the engine instruments in the center of the instrument panel and is controlled by a rheostat, marked NON-FLIGHT INST LIGHTS, OFF, BRT, located to the left of the flight instrument rheostat. The intensity of the lights is controlled by rotating the rheostats.

Figure 4—11. Landing Light Switches

OVERHEAD PANEL, QUADRANT, AND CONSOLE LIGHT RHEOSTAT.

The panel lights on the three overhead panels, the engine control quadrant, the radio and interphone control panels, the landing gear control panel, and the check lists are controlled by a rheostat, marked CONSOLE & PANEL LIGHTS, OFF, and BRT, located to the right of the instrument light rheostats. The intensity of the lights may be varied by rotating the rheostat.

PILOT'S AND COPILOT'S SPOTLIGHTS.

The pilot's and copilot's spotlights (4 and 1, figure 1—7) are mounted in adjustable brackets, one on each side of the overhead switch panel. The spotlights are equipped with dimming rheostats and removable red filters.

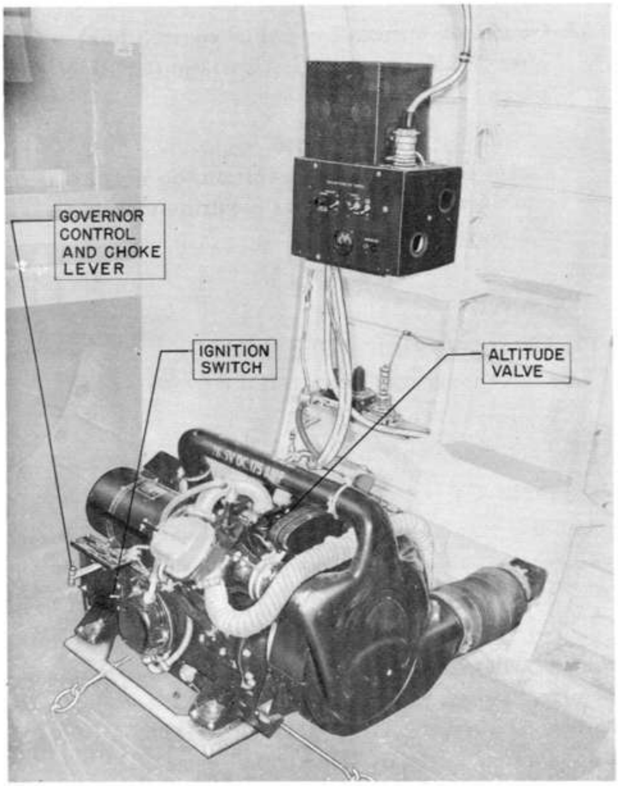


Figure 4—12. Auxiliary Power Unit

CABIN DOME LIGHT SWITCH.

A guarded cabin dome light switch (figure 4—2), marked DOME LIGHTS, CABIN, located on the pilot's compartment dome light panel, controls six dome lights located in the cabin ceiling. Each light contains a red and white lamp. The switch has three marked positions: R, OFF, and W. To turn on the red lamps, place the switch in the R position; to turn on the white lamps, open the guard and place the switch in the W position.

AUXILIARY POWER UNIT.

An auxiliary electric power system, consisting of a portable auxiliary power unit (16, figure 1—3) and an auxiliary power unit control box (figure 4—13), is used primarily to operate the cargo hoist, though it can be used to start the engines and to deliver power for the operation of any of the electrical equipment installed in the helicopter. The unit is installed in the aft left corner of the cabin and is fastened to cargo tie-down rings in the floor by four wing nuts. The portable APU (figure 4—12) consists of a combination starter-generator and a two-cylinder, four-cycle gasoline engine that drives the generator after being started. The auxiliary system connects to the primary bus, both to receive battery power for starting the APU and to deliver APU output to the helicopter electrical distribution system. The engine receives fuel from the left fuel tank through a pump and

Figure 4—13. Auxiliary Power Unit Control Box

line that also supplies the fuel for the cabin heater. The fuel line leading from the left fuel tank to the heater and APU fuel pump is equipped with an APU and heater fuel manual shut-off valve accessible through a removable panel on the lower inboard side of the left nacelle. This valve must be in the open position (handle vertical) before the heater or APU can obtain fuel. Fuel shut-off solenoid valves are located on the cabin side panel behind the power unit. The power unit engine exhaust and cooling air intake are ducted through the left side of the cabin. A starter cord, stowed in a service kit located at the rear of the unit, may also be used for starting the engine. The auxiliary power unit control box is located above the power unit on the cabin side panel and contains the control switches, circuit breakers, and relays that connect the auxiliary power generator voltage to the primary bus. The power unit with its control box and power cable may be removed from the helicopter to reduce weight or to make room for cargo or the 36-troop seat installation.

APU FUEL PUMP SWITCH.

The circuit breaker switch, marked FUEL PUMP and OFF, located on the auxiliary power unit control box, controls the fuel pump in the left engine nacelle and fuel solenoid valve at the APU. Placing the switch in the FUEL PUMP position turns on the fuel pump and opens the solenoid valve, permitting fuel to flow to the carburetor of the APU. Placing the switch in the OFF position turns off the pump and closes the valve.

APU STARTER SWITCH.

The APU starter switch, located on the auxiliary power unit control box, marked START and RUN, is used to start the auxiliary gasoline engine. When the switch is held in the START position, direct current from the primary bus will motorize the APU generator and start the engine. The switch is spring-loaded and will return to the RUN position when finger pressure is released.

APU GENERATOR SWITCH.

The left switch on the auxiliary power unit control box is marked GENERATOR and has two marked positions, ON and OFF. After the engine is running, the APU generator switch is turned to the ON position to connect the generator output to the helicopter primary bus. A reduction in engine speed may be anticipated when electric load is applied.

APU IGNITION SWITCH.

The toggle switch (figure 4—12), marked IGNITION, has two marked positions, OFF and ON. The switch, located on the panel at the lower left-hand side of the auxiliary power unit, connects the magneto to the spark plugs of the APU.

APU ALTITUDE VALVE.

A thumb screw (figure 4—12), located on the carburetor between the cylinders of the APU, must be adjusted when the APU is used at altitudes above sea level. The pointer should be set opposite the figure nearest to the altitude at which the engine is to be operated.

APU GOVERNOR CONTROL AND CHOKE LEVER.

At the left-hand side of the APU is a lever (figure 4—12), working in a horizontal quadrant, marked CHOKE, IDLE, and RUN. The lever is placed in the CHOKE position to aid in starting at low temperatures, in the IDLE position during warm-up, and in the RUN position after the engine has warmed up.

APU GENERATOR FAILURE WARNING LIGHT.

A red press-to-test warning light, located to the right of the generator switch, will light if there is no generator output.

APU AMMETER.

Below the switches on the auxiliary power unit control box is an ammeter which indicates the amount of current being drawn from the APU. The ammeter should not indicate a continuous load of more than 175 amperes, though for periods no longer than 5 minutes, the load may be as high as 263 amperes.

NORMAL OPERATION.

STARTING.

1. Battery switch (in pilot's compartment) — ON.
2. Fuel pump switch (on APU control box) — FUEL PUMP.
3. Ignition switch (on APU) — ON.
4. Governor control lever (on APU) — IDLE.
5. Starter switch (on APU control box) — START.
6. Governor control lever — CHOKE (if necessary) for starting. Move to the IDLE position until engine warms up, and then to the RUN position for 2 minutes without load.
7. Generator switch (on APU control box) — ON, after 2 minutes of engine operation at RUN position.

Note

Except for momentary operation, do not use a combination of equipment requiring more than 175 amps.

STOPPING.

1. Generator switch — OFF.
2. Governor control lever — IDLE for 5 minutes.
3. Ignition switch — OFF.
4. Fuel pump switch — OFF.

EMERGENCY OPERATION.

The APU may be started manually by winding the starter cord around the grooved pulley located on the flywheel and pulling from any convenient angle. More choking will be required.

CARGO COMPARTMENT.

CABIN.

The fuselage of this helicopter contains a cargo compartment 30 feet 4 inches long, 7 feet 8 inches wide, and 6 feet 8 inches high. At the forward end of the compartment under the radio deck, vertical clearance tapers down to approximately 5 feet when the ramp is raised. Approximately 1520 cubic feet of space is available for cargo stowage. The only obstruction in this space is the removable auxiliary power unit (16, figure 1—3), located on the floor in the aft left corner of the compartment, and the pilot's compartment ladder (8, figure 1—3), located in the center forward section. If necessary, this ladder may be folded upward and the pilot and copilot can enter or leave the helicopter through the pilot's compartment right sliding windows. The cargo compartment floor is strongly reinforced, especially in two lengthwise strips leading up the nose ramp and through the cabin in order to accommodate the wheels of various mobile units. The floor regions less able to accommodate a concentrated load, are the battery cover aft of the large central hatchway, the cargo hatch cover (9, figure 4—14) at the forward end of the hatch, and the two floor segments attached to the nose doors. Floor loading should not exceed 300 pounds-per-square-foot. For loading purposes, the cabin is divided into eight regions fore-and-aft. Decals on the cabin side panels state the maximum load and the center-of-gravity location for each region. For maximum permissible weight to be carried in the various compartments, refer to AN 01-1B-40, Handbook of Weight and Balance, revised Chart E, chart date 6 March 1958. There are 112 cargo tie-down rings (2, figure 4—14), loocated in the cargo compartment floor. Nine of the rings along each outboard side of the floor are heavier

1. Cargo Hoist Monorail
2. Cargo Tie-Down Rings
3. Cargo Tie-Down Strap Stowage Pocket (Typical)
4. Nose Doors Open
5. Nose Door Manual Catch
6. Ramp
7. Ramp Extensions
8. Cargo Release Handle Access Door
9. Cargo Hatch Cover
10. Cargo Sling (Stowed Position)
11. Cargo Sling Stowage Line
12. Cargo Door
13. Cargo Hoist
14. Monorail Supports

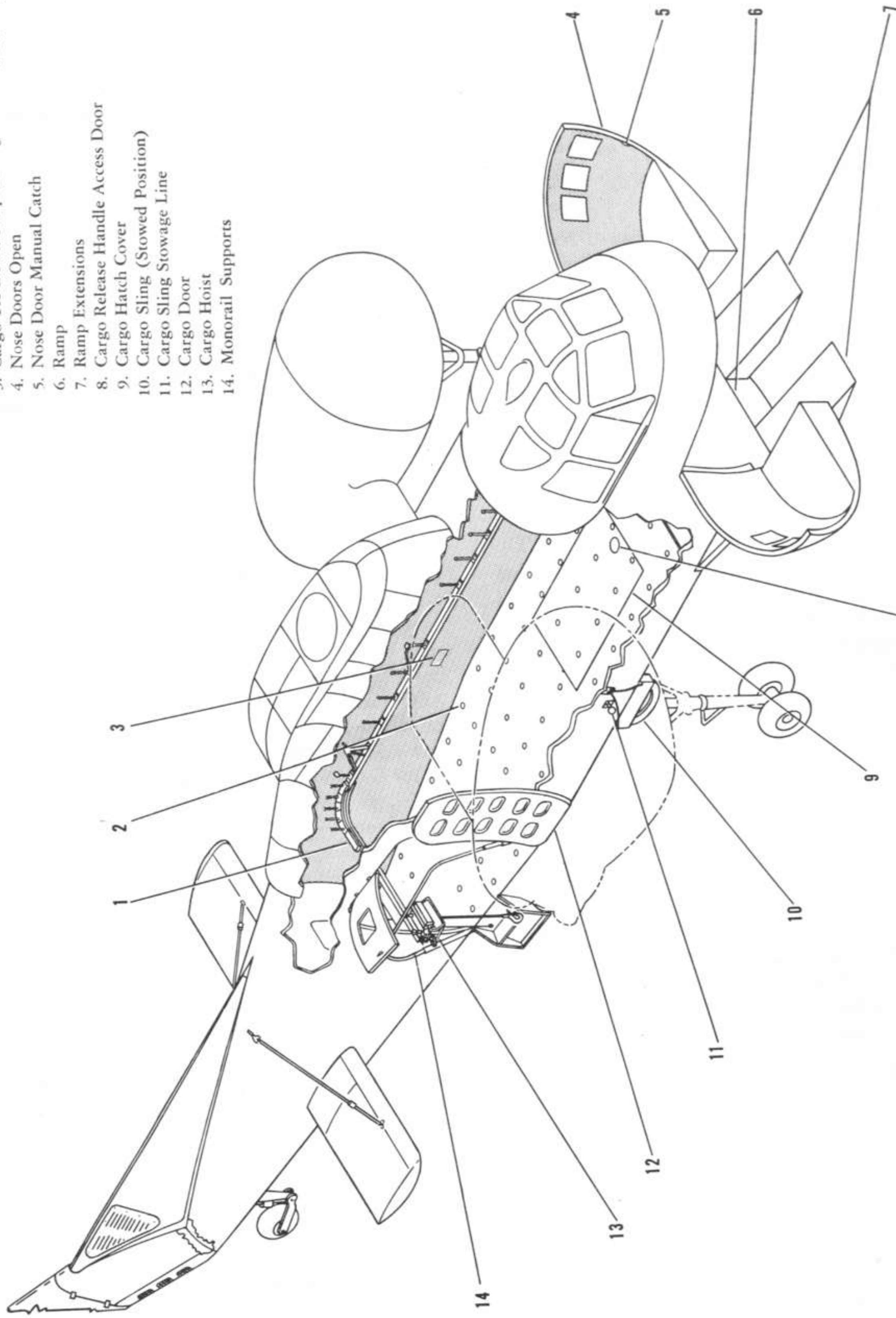


Figure 4–14. Cargo Handling Facilities

and can take a load of 5000 pounds. The rest of the rings are capable of taking 2200 pounds. Thirty 5000-pound fabric cargo tie-down straps are stowed in cargo tie-down strap stowage pockets (3, figure 4–14), located along the cabin side panels. Cabin personnel must be seated aft of the cargo when both are carried. When loading the helicopter, refer to Section V for center of gravity limitations and maximum recommended gross weight, to the Load Adjuster, R1680-953-3691-GA20, and to AN 01-1B-40, Handbook of Weight and Balance.

CARGO DOOR.

A cargo door (12, figure 4–14), approximately 6 feet square, is located at the aft end of the cabin on the right side of the fuselage. The aft half of the cargo door is a passenger door (26, figure 1–2) composed of two panels. The upper panel is unlocked and swung outward and upward from outside or inside of the helicopter by means of handles at its lower edge. The upper door panel is supported in the open position by struts at its forward and aft edges. The lower panel of the passenger door is unlocked and swung outward and downward to form a step, by means of a handle located on the outside at its upper edge. The forward half of the cargo door is unlocked and swung outward and forward by means of two latches located inside the door at the top and bottom. When opening the cargo door, open first the upper panel, then the lower panel, and lastly the forward half. Close the doors in the reverse sequence. Do not open any of the cargo door panels in flight except the upper panel which may be jettisoned from inside in case of emergency.

CARGO HATCH.

A 6- by 4-foot hatch, in the center of the cabin floor, may be used for cargo or rescue operations. The skin panels on the bottom of the fuselage that cover the hatchway and the hatch cover, flush with the cabin floor, are removed before take-off. Personnel in the cabin should wear safety belts attached to the cargo tie-down rings whenever flights are made with the cargo hatch open.

NOSE DOORS AND RAMP.

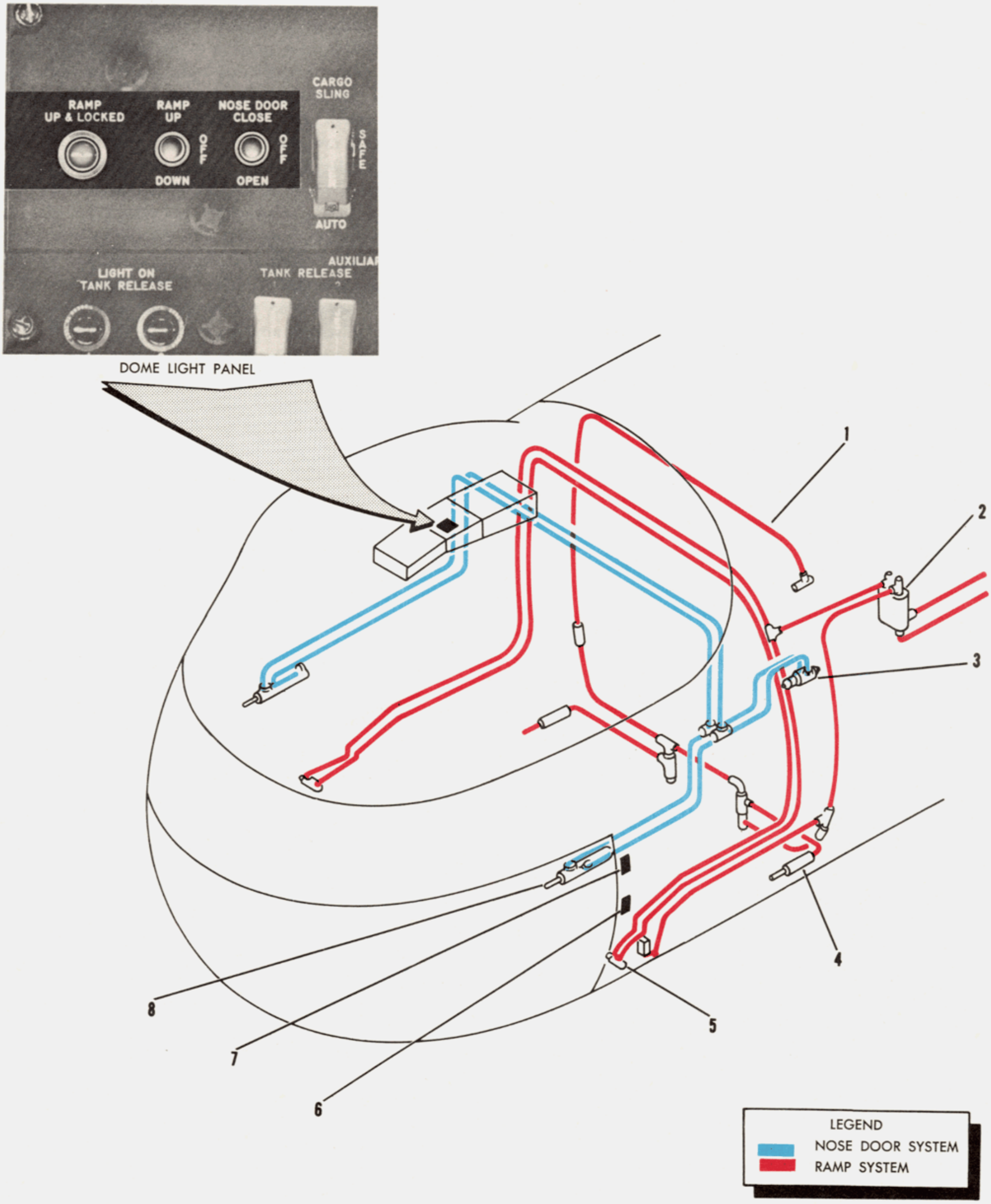
Two clamshell nose doors (4, figure 4–14), open outward to permit loading through the forward part of the cargo compartment. The doors are opened and closed hydraulically by electric control switches in the pilot's compartment. The maximum forward speed, with clamshell nose doors open, is 40 knots CAS. A lock lever on the control console must be released before hydraulic pressure will actuate pistons for opening and closing the doors. Prior to actuating the nose door controls, a nose door manual catch (figure 4–17), installed on the inside of the doors, just above the ramp, must be unlocked. After closing the nose doors, the catch must be locked. After the nose doors have been opened, a ramp (6, figure 4–14), consisting of a portion of the cabin floor aft of the door opening, may be lowered until its forward end contacts the ground. At the forward end of the ramp are ramp extensions (7, figure 4–14) which stand vertically and are secured with swivel hooks to the sides of the cabin in their stowed positions. After the ramp has been lowered, the ramp extensions are unhooked and swung downward to permit vehicles to be driven onto the ramp and into the cabin. The clearance to the flight deck structure perpendicular to the ramp surface is 6 feet 8 inches. The maximum weight that may be stowed on the ramp and still permit the ramp to be raised, is approximately 2000 pounds. The ramp is raised by hydraulic power from the utility hydraulic system and lowered by gravity with hydraulic damping. Ramp controls, consisting of a switch and warning light, are located on the pilot's compartment dome light panel. The ramp is automatically locked in the up position when raising is completed. The ramp up-lock cylinders, located behind small doors on the inside of each fuselage side panel at the forward end of the ramp, may be operated manually by pulling aft on emergency ramp lock levers to unlock the ramp in case of hydraulic failure. Before opening the nose doors and lowering the ramp, be sure that the pilot's compartment ladder is raised against the cabin overhead.

Note

The forward end of the ramp contacts the ground at two points directly under the reinforced "roadway" strips. Before lowering the ramp, make sure that the ground under these points is level and of equal load-carrying ability to avoid twisting the ramp or the ramp extensions when loads are applied. Make sure the ground is free of rocks, stumps, etc, aft of the load points to avoid damaging the ramp lower skin.

NOSE DOOR LOCK LEVER.

The nose door lock lever (figure 4–16), located on the left side of the control console, manually controls the nose door lock. The lever moves fore-and-aft in a quadrant, marked NOSE DOOR, and has two marked positions, UNLOCKED and LOCKED. Mechanical linkage leads from the lever down to the lock, which is located above the doors on the center line of the fuselage. The nose door lock lever must be placed forward in the UNLOCKED position before nose door opening or closing and ramp actuating can be accomplished. A latch, to retain the nose door lock lever in the LOCKED position, must be pushed inboard before the lever can be moved. In high wind conditions, the nose door switch must be placed in the closed position prior to releasing the nose door lock lever.



1. Emergency Hydraulic Line (BuNos 138418 through 138424)
2. Ramp Control Valve
3. Nose Door Control Valve
4. Ramp Actuating Cylinder
5. Ramp Latch Actuating Cylinder
6. Ramp Up-Lock Limit Switch
7. Nose Door Limit Switch
8. Nose Door Actuating Cylinder

Figure 4–15. Nose Door and Ramp System

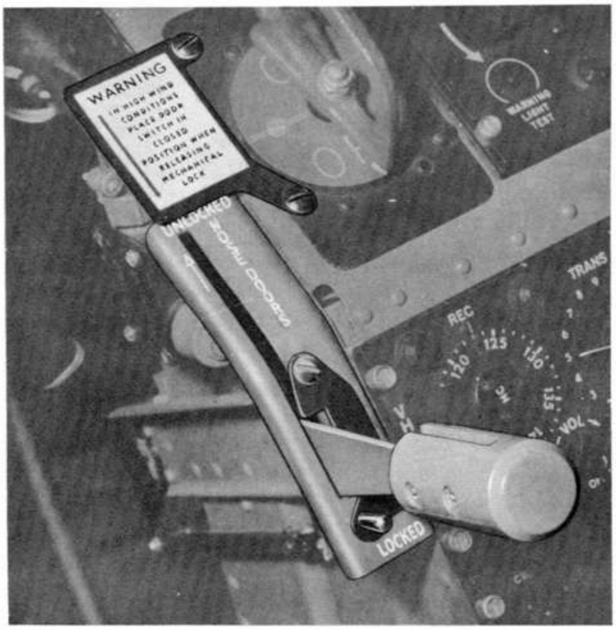


Figure 4–16. Nose Door Lock Lever

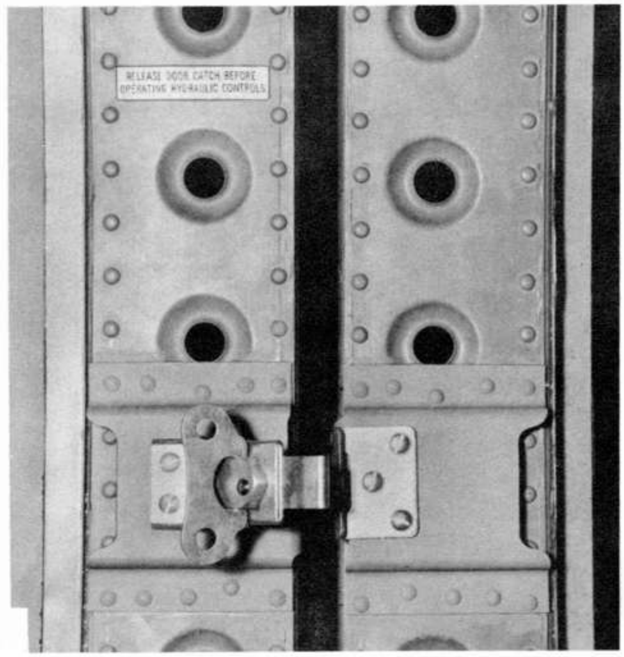


Figure 4–17. Nose Door Manual Catch

NOSE DOOR SWITCH.

A nose door switch (figure 4–2), marked NOSE DOOR, with three marked positions, CLOSE, OFF, and OPEN, located on the left side of the pilot's compartment dome light panel, controls the actuation of the nose doors through direct current from the secondary bus. The switch is inoperative until the nose door lock lever has been manually released. When the switch is placed in the CLOSE or OPEN position, the doors are actuated. The switch is returned to the OFF position at the completion of either operation. Do not close the nose doors unless the ramp is up.

RAMP SWITCH AND INDICATOR LIGHT.

The ramp switch (figure 4–2) marked RAMP, located to the left of the nose door switch, has three marked positions: UP, OFF, and DN. The switch controls the hydraulic ramp actuation by direct current from the secondary bus. The switch is inoperative until the nose doors have been opened. When the switch is placed in the UP or DN position, the ramp is actuated. The switch is returned to the OFF position at the completion of either operation. To the left of the ramp switch is a green indicator light, marked RAMP UP & LOCKED, which will go on when the ramp is locked. The ramp switch is then turned OFF and the light will go out.

NOSE DOOR AND RAMP HYDRAULIC VALVES.

The electrically operated nose door and ramp hydraulic valves (figure 4–18) that control actuation of the nose doors and ramp are located on the left-hand cabin side panel, approximately under the main gear box. The forward valve controls the ramp and the aft valve, equipped with a hinged cover to protect it from heat from the register directly above, controls the nose doors. In case of electrical failure, these valves must be positioned manually in order to actuate the nose doors and ramp. Each valve has two spring-loaded manual override knobs, one at each end, and is placarded to indicate which knob must be held in to accomplish the actuation required. The knobs should never be actuated while the helicopter is airborne. Each knob is equipped with a removable safety collar that clamps around its stem to prevent accidental actuation. If the override knobs have been actuated, be sure that the collars are replaced immediately.

NOSE DOOR OPEN LOCK BOLT.

The nose door manual override knob is equipped with a nose door open lock bolt which is slid aft to hold the door open manual override knob in the actuated position in order to prevent the doors from swinging closed during cargo loading. The lock bolt is operative whether the nose door valve has been actuated electrically or manually. After the lock bolt is actuated, the helicopter electric system may be shut down and the doors will remain open. To actuate the lock bolt, rotate it upward to clear the stowed position detent, slide it aft to actuate the override knob and rotate it downward into the locked position detent. Be sure the lock bolt is released and secured in the forward detent before attempting to close the nose doors electrically or manually.

EMERGENCY RAMP LOCK LEVERS.

In case of utility system hydraulic failure, the ramp may

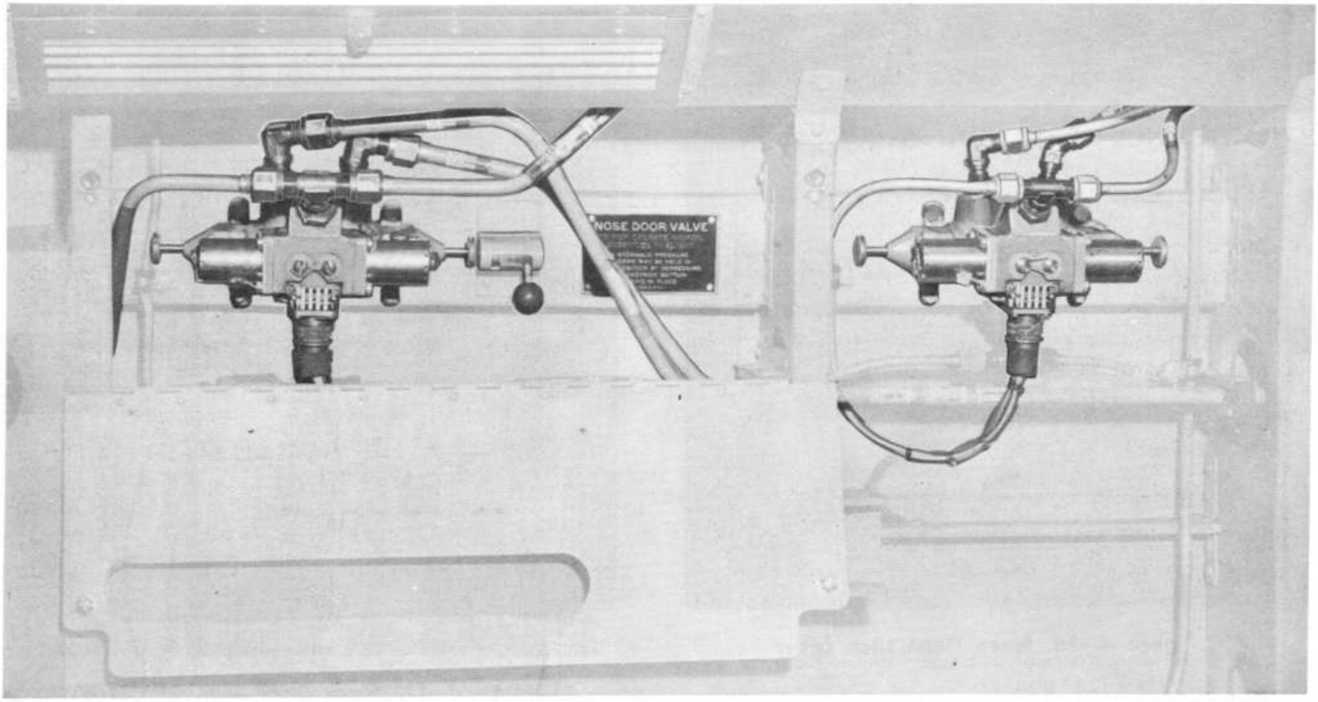


Figure 4—18. Nose Door and Ramp Hydraulic Valves

be unlocked by pulling aft on two levers located one on each side, behind the small doors in the lower forward corners of the fuselage side panels. The doors are hinged at the bottom and are held closed by two fasteners at the top. Both levers must be actuated before the ramp will be released to lower under its own weight.

NORMAL OPERATION.

The left engine should be running while operating the nose doors and ramp in order to maintain hydraulic pressure in the utility system. Operation may also be accomplished by using external hydraulic and electric power.

To open nose doors and lower ramp:

1. Left control quadrant throttle – Adjust to 1200 to 1500 engine rpm.
2. Pilot's compartment ladder – Folded up.
3. Nose door manual catch – Release.

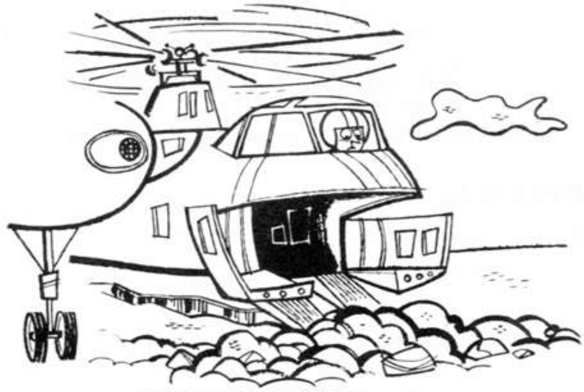
Note

In high wind conditions, place the door switch at CLOSE prior to releasing manual locks.

4. Nose door lock lever – UNLOCKED.
5. Nose door switch – OPEN, then OFF when doors are open.
6. Ramp switch – DN, then OFF when ramp is down.
7. Ramp extensions – Manually unhook and fold down.
8. Nose door open lock bolt – Lock, to prevent doors from closing.

To raise ramp and close nose doors:

1. Left control quadrant throttle – Adjust to 1200 to 1500 engine rpm.
2. Ramp extensions – Manually fold up and secure with hooks.
3. Ramp switch – UP, then OFF when ramp up warning light comes on.
4. Nose door open lock bolt – Release.
5. Nose door lock lever – UNLOCKED.
6. Nose door switch – CLOSE, then OFF when doors are closed.



THIS THING AIN'T NO BULLDOZER!! RETRACT THE RAMP AND CLOSE THE NOSE DOORS BEFORE TAXIING.

7. Nose door lock lever – LOCKED.

8. Nose door manual catch – Latch and lock.

EMERGENCY OPERATION.

ELECTRICAL FAILURE.

In case of electrical failure, the nose door and ramp hydraulic valves (figure 4–18), located on the left cabin side panel forward, may be positioned manually. Since the interlock circuits and the ramp-up indicator light will also be inoperative, be sure each operation is completed before proceeding to the next.

To open nose doors and lower ramp:

1. Left engine quadrant throttle – Adjust to 1200 to 1500 rpm.
2. Nose door manual catch – Release.
3. Nose door lock lever – UNLOCKED.
4. Nose door manual override knob – Remove safety collar, OPEN, then OFF when doors are open.
5. Nose door open lock bolt – Lock.
6. Ramp manual override knob – Remove safety collar, LOWER, then OFF when ramp is down.
7. Ramp extensions – Unhook and fold down.

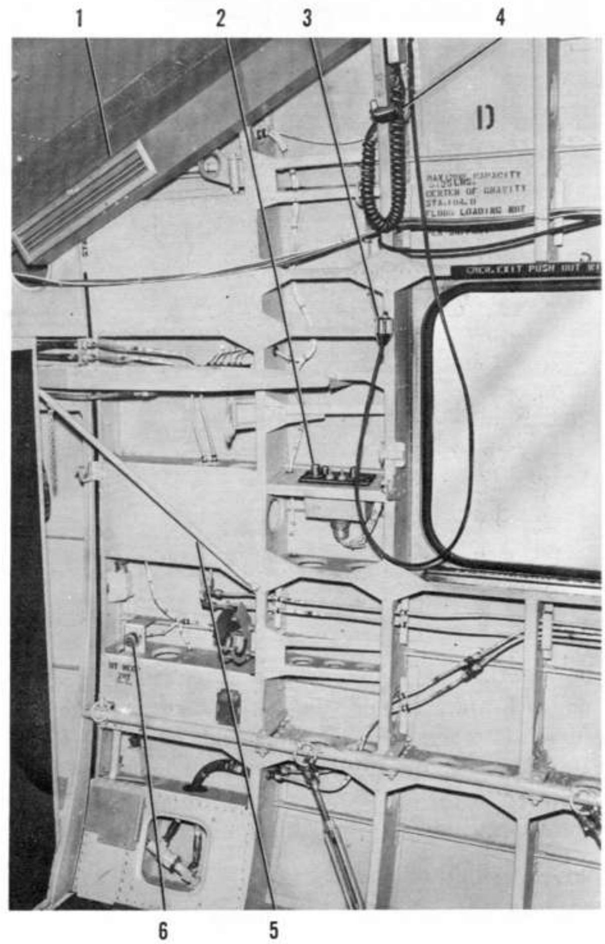
To raise ramp and close nose doors:

1. Left engine quadrant throttle – Adjust to 1200 to 1500 rpm.
2. Ramp extensions – Fold up and secure.
3. Ramp manual override knob – RAISE, then OFF when ramp is up and locked. Replace safety collar.
4. Nose door lock lever – UNLOCKED.
5. Nose door open lock bolt – Release.
6. Nose door manual override knob – CLOSE, then OFF when doors are closed. Replace safety collar.
7. Nose door lock lever – LOCKED.
8. Nose door manual catch – Latch and lock.

HYDRAULIC FAILURE.

To open nose doors and lower ramp:

1. Nose door manual catch – Release.
2. Nose door lock lever – UNLOCKED.
3. Swing doors open manually.
4. Nose door open lock bolt – Lock.



1. Cabin Register
2. ICS Control Panel
3. Portable Microphone Switch
4. Headset Connection Box
5. Ramp Extension Swivel Hook
6. Utility Receptacles

Figure 4–19. Forward Crew Station

5. Ramp manual override knob – Remove safety collar, then LOWER.
6. Emergency ramp lock levers – Pull aft; ramp will lower slowly.
7. Ramp extensions – Unhook and fold down.

To raise ramp and close nose doors:

1. Ramp extensions – Fold up and secure.
2. (Deleted).
3. (Deleted).
4. (Deleted).
5. (Deleted).

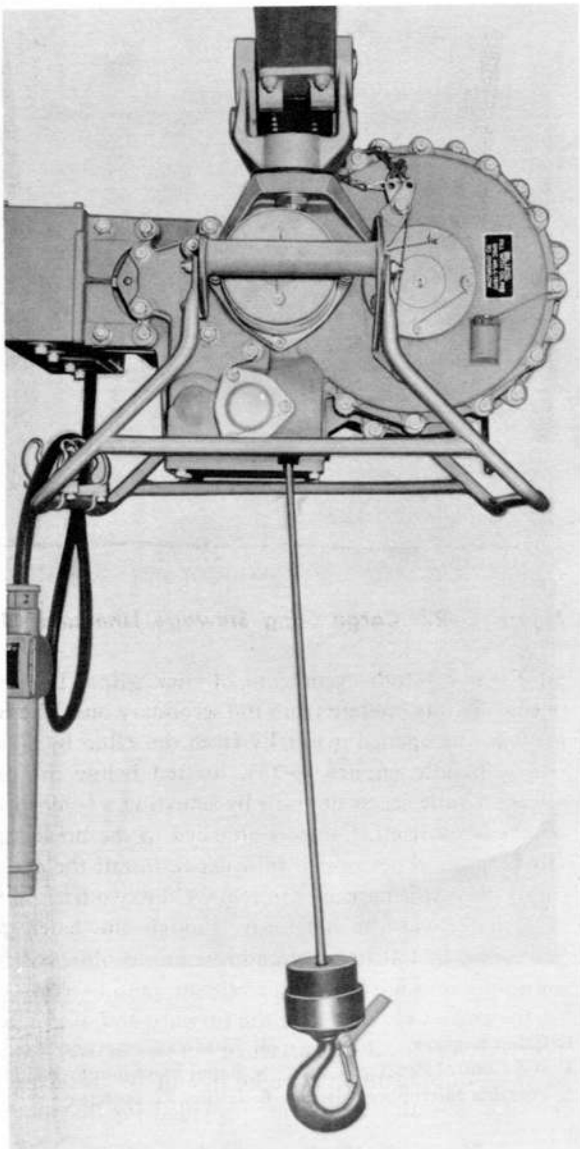


Figure 4–20. Cargo Hoist

6. (Deleted).
7. (Deleted).
8. Nose door open lock bolt – Release.
9. Swing doors closed manually.
10. Nose door lock lever – LOCKED.
11. Nose door manual catch – Latch and lock.
12. Emergency hydraulic valve – Open.

Note

It is necessary to manually lift the ramp. When the ramp is fully raised, push the emergency ramp lock levers forward.

CARGO HOIST.

A cargo hoist (13, figure 4–14), capable of lifting 2000 pounds, travels on a cargo hoist monorail (1, figure 4–14), suspended from the middle of the cabin overhead. The cargo hoist consists of an electrically operated, motor-driven winch, which drives a drum containing 50 feet of cable, and four trolley wheels which travel on the rail. A slip clutch is provided in the winch mechanism to limit the lifting capacity to 2000 pounds, thereby preventing the possibility of an overloaded cargo being lifted. The winch is equipped with three locks to secure it to the rail at various points. Two of the locks consist of removable pins located below each set of trolley wheels. These lockpins are used to secure the winch to the rail for performing rescue operations through the hatchway in the cabin floor. A rescue sling may be attached to the cable hook for this purpose. Maximum capacity for in-flight rescue operations is 600 pounds. The lockpins are also used to secure the winch to the specially braced short portion of the rail just forward of the curved portion for dragging wheeled equipment up the nose ramp. This is the only point from which dragging may be accomplished and it is stenciled to indicate this limitation. Dragging is restricted to 2000 pounds maximum and must be carried out parallel to and along the center line of the helicopter. It is at this point that the winch is to be secured for flight. The third lock consists of a spring-loaded pin operating through the hub of one set of trolley wheels. This pin extends through holes in the web of the rail to anchor the winch at various points for the depositing of cargo picked up from the rear cargo door. The auxiliary power unit is the normal source of electric power for the operation of the hoist so that cargo may be handled without the engines running. The winch electric power cable is connected to the secondary bus at a receptacle on the right cabin bulkhead. The monorail extends from the front of the cabin to a point opposite the aft cargo door where it curves 90 degrees and extends out the door, being supported at its outer end by two monorail supports (14, figure 4–14) which are attached to the fuselage bottom structure with quick-disconnect pins. When the monorail is rigged through the doorway, the struts that support the passenger door must be disconnected from the fuselage and re-attached at a higher point to allow the door to clear the rail. The curved portion of the monorail and the two supports must be removed in order to close the cargo door prior to take-off. A short portion of the monorail just forward of the curve is well braced to the cabin overhead and is the only point from which dragging may be accomplished. The monorail may be folded up to the right and locked against the overhead to increase the head

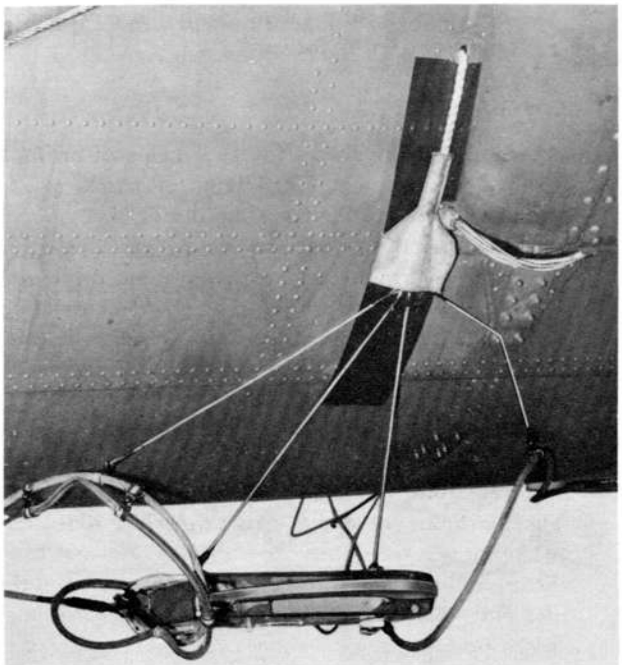


Figure 4–21. Cargo Sling – Stowed Position

room in the cabin or it may be removed from the helicopter to save weight.

CARGO HOIST TWO-SPEED WINCH.

On BuNos 140314 and subsequent, a two-speed cargo hoist winch is installed. The winch has a gear shift installed on the forward end with high and low speed positions appropriately marked on the winch casting. Loads up to 600 pounds may be hoisted at an accelerated rate which allows faster pick-ups during rescue operations. The low speed feature is still retained for heavy hoisting and cargo winching.

CARGO HOIST CONTROL SWITCH.

The cargo hoist winch is controlled by a cargo hoist control switch, located on the 4-foot length of cable attached to the hoist. The switch has two marked positions: UP and DOWN. Whenever the switch is not actuated, an automatic brake will hold the load. To actuate the winch, push the thumb switch to either extreme position. The hoist operates on direct current from the secondary bus.

CARGO SLING.

An external cargo sling (10, figure 4–14), capable of carrying 10000 pounds, is attached to the four corners of the floor hatchway. The four cables extend from the bottom of the fuselage to a hook assembly. The cargo sling hook may be opened electrically to release an external load by a solenoid, either automatically when the load is resting on the ground and the cargo attaching cables are slack, or by depressing thumb switches on the pilot's or copilot's cyclic control stick grips. The cargo release circuits operate from the secondary bus. The hook may also be opened manually from the cabin by a cargo release handle (figure 4–23), loacted below the cargo release handle access door, or by actuating a lever located on the hook itself. Cargo is attached to the hook manually by ground personnel. In order to install the external cargo sling, it is necessary to remove the two bottom fairings that cover the hatchway, though the hatch floor panel may be left in place and operations observed from the cabin through the cargo release handle access door (8, figure 4–14), located at the forward end of the hatch. When the cargo sling is attached but not in use, it should be drawn up against the right side of the fuselage into the stowed position by a line provided for that purpose.

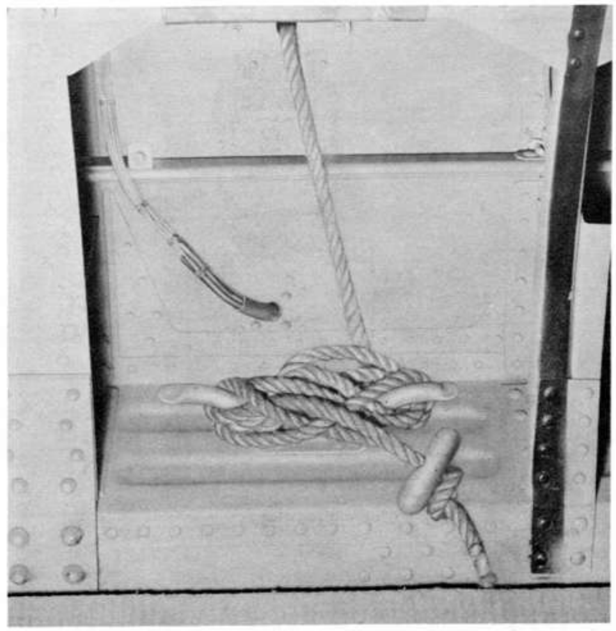


Figure 4–22. Cargo Sling Stowage Line and Cleat

CAUTION

To avoid damaging the cargo sling hook, and to avoid danger of hooking onto ground obstructions, the cargo sling should be stowed at all times except when actually handling cargo. After releasing a load from the cargo sling in flight, stow the sling before landing.

CARGO SLING MASTER SWITCH.

A guarded cargo sling master switch (figure 4–2), marked CARGO SLING MASTER, located on the pilot's compartment dome light panel, controls the operation of the cargo sling hook. The switch has three marked positions: SAFE, SLING, and AUTO. The switch is normally left in the SAFE position and the hook may be closed or

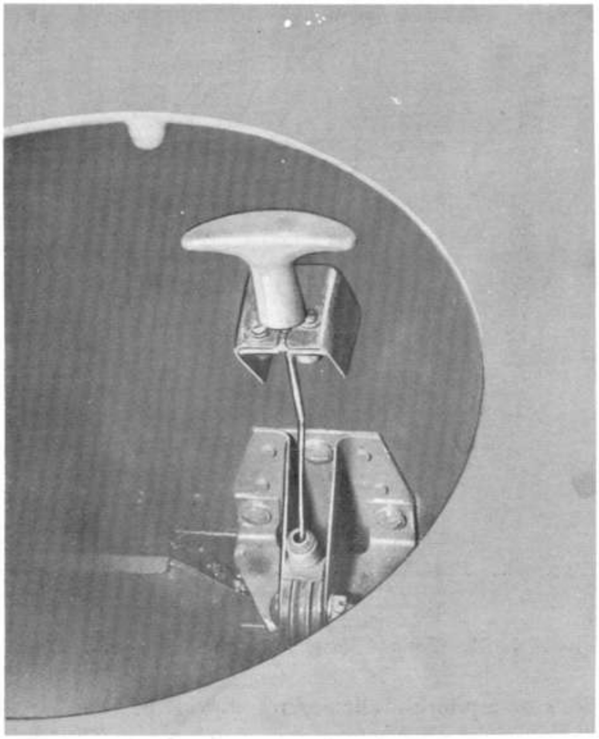


Figure 4—23. Cargo Release Handle

opened by ground personnel or opened by pulling the handle in the hatchway. When preparing to release cargo electrically, the switch is placed either in the SLING or AUTO position. With the switch in the SLING position the cargo is dropped by actuating either cyclic control stick thumb switch. When the switch is placed in the AUTO position, the cargo will be released automatically when the load contacts the ground and the weight of the load on the hook is reduced to approximately 120 pounds. Whenever the load on the hook is 120 pounds or less, the hook will open. The hook release switch is set to operate at this value as this is the maximum estimated weight of slings or nets used to attach the cargo load to the hook. The thumb switches on the grips of each cyclic control stick may also be used to release the load when the master switch is in the AUTO position. The switch should be returned to the SAFE position and the sling stowed after dropping cargo.

CAUTION

When carrying loads of less than 150 pounds, the cargo sling master switch should never be placed in the AUTO position. The cargo sling hook would open immediately as if a heavier load were resting on the ground. To release a load of less than 150 pounds, place the master switch in the SLING position and actuate the cyclic control stick thumb switch when the drop is desired.

ALWAYS STOW THE CARGO SLING BEFORE LANDING.

CARGO RELEASE SWITCHES.

A cargo release switch (figure 1—28), marked CARGO RELEASE, is located on the pilot's and copilot's cyclic control stick grips. Either switch may be depressed to open the cargo sling hook when the cargo sling master switch is in the SLING or AUTO position. The thumb switches must be used to release loads of less than 150 pounds with the master switch placed in the SLING position.

CARGO RELEASE HANDLE.

A manual cargo release handle (figure 4—23) is located under the cargo release handle access door at the forward end of the cabin floor hatch. The access door is opened and the handle is pulled up to release the cargo. A spring returns the hook lock and the handle to the original position.

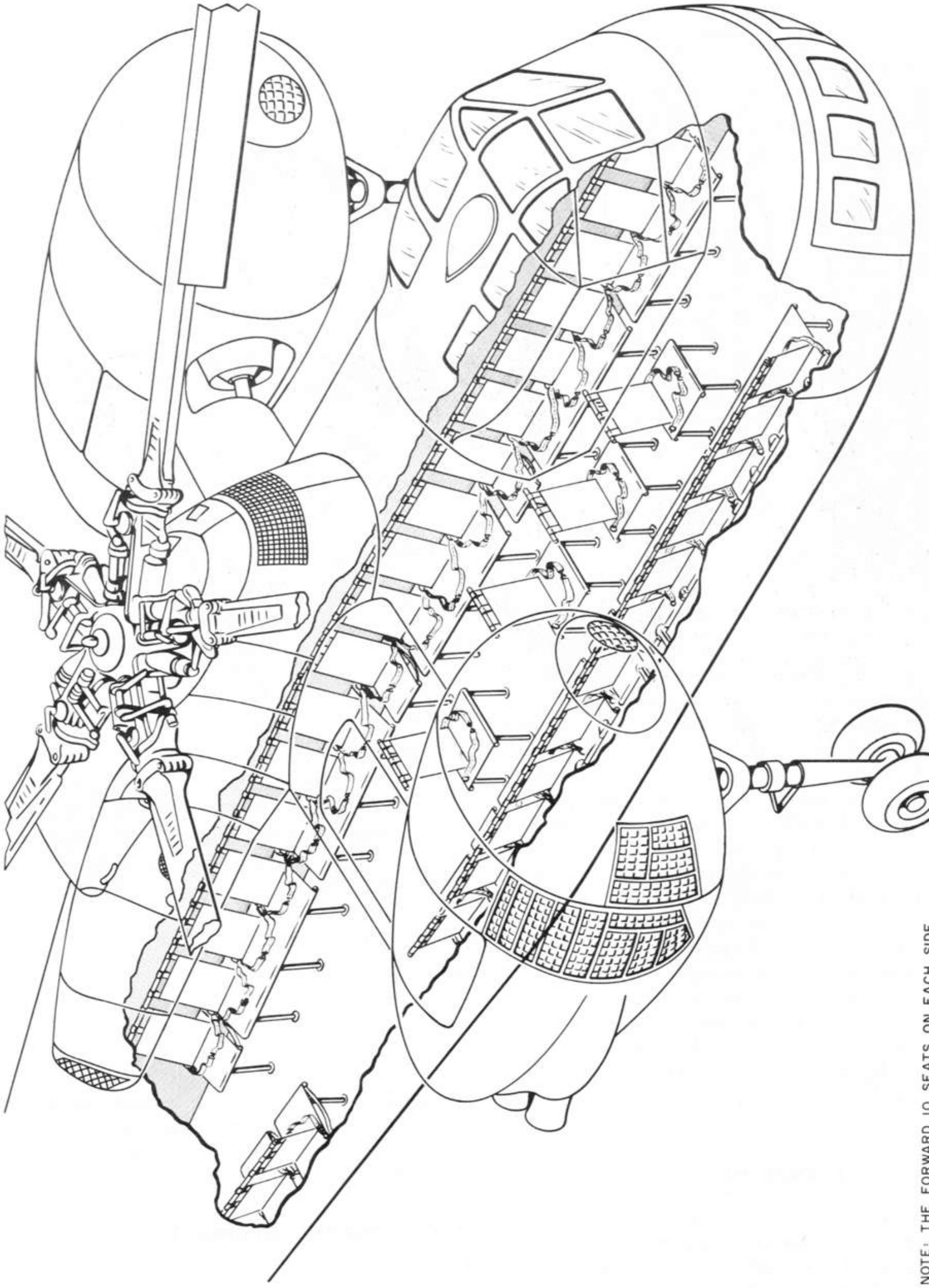
CARGO SLING STOWAGE LINE.

The cargo sling stowage line (figure 4—22), attached to the cargo sling hook, passes into the cabin on the right side forward of the external power receptacle and is secured to a cleat located on the cabin floor between two frames of the right-hand side panel. To stow the cargo sling, pull in on the line and secure it to the cleat. To unlatch the cargo sling from its stowed position, grasp the standing part of the line (to prevent the sling hook from falling to the ground), cast it off from the cleat, and pay out.

TROOP CARRYING EQUIPMENT.

TROOP SEATS.

Folding troop seats (figure 4—24), equipped with safety belts, may be installed in the cabin to accommodate either 20 or 36 fully-equipped troops. The seat backs,



NOTE: THE FORWARD 10 SEATS ON EACH SIDE COMPRISE THE 20 TROOP SEAT INSTALLATION.

Figure 4–24. Troop Seat Installation

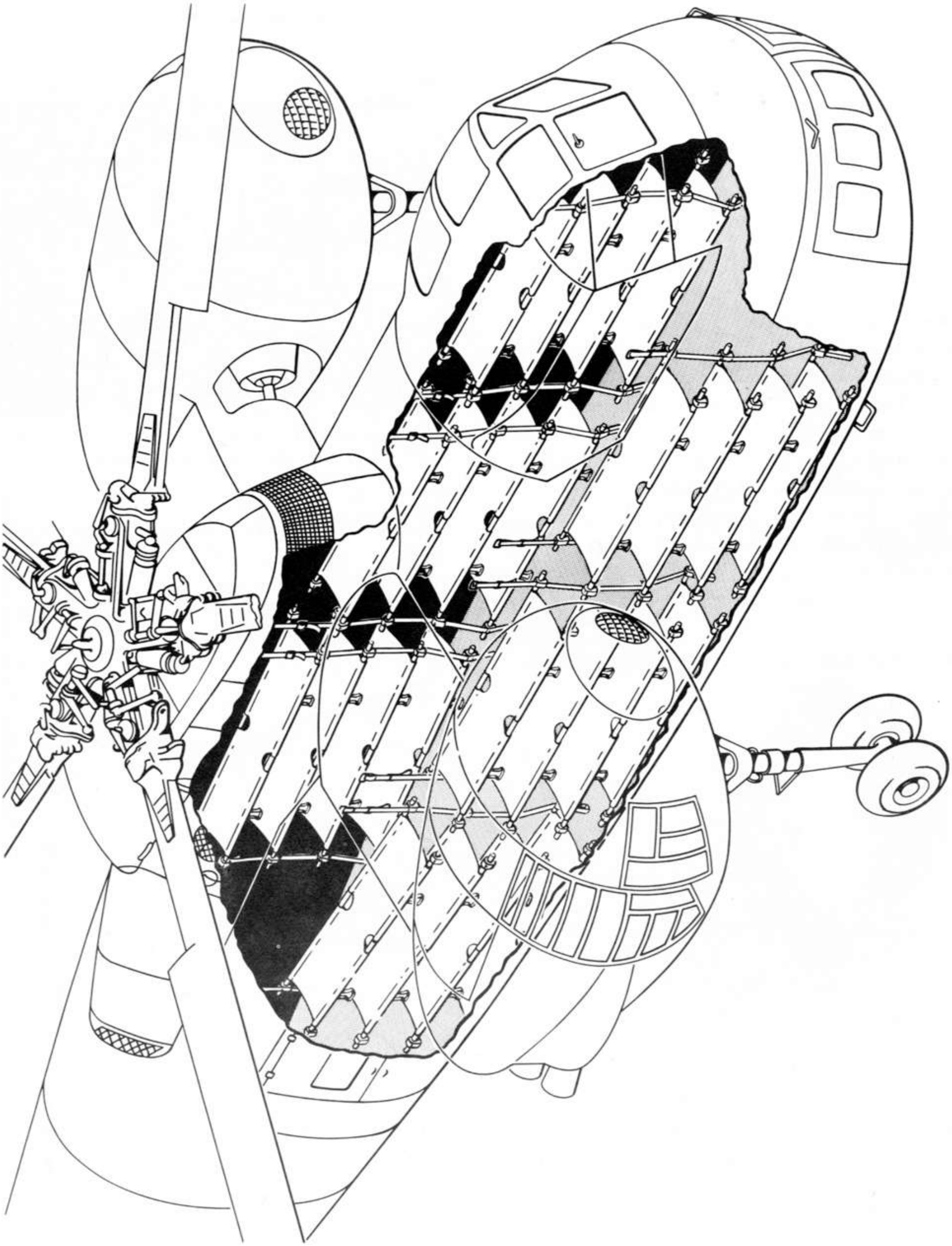


Figure 4—25. Litter Installation

Figure 4—26 deleted.

in one-man and three-man assemblies, are attached to each cabin side panel facing inboard. The seats are folded by disconnecting the front legs from the floor and securing the front of the seats against the upper back support bar with the straps provided. In the 20-seat configuration, 10 seats are installed on each side at the front of the cabin. To increase the number of seats to 36, 16 seats are added; eight on the left side, four on the right, and four individual seats facing forward in the center. To carry the extra troops, it is necessary to remove the auxiliary power unit, its electrical control box, and power cable from the cabin. Personnel must be seated aft of the cargo when both are carried.

CASUALTY CARRYING EQUIPMENT.

Provisions are made in the cabin for the installation of 24 pole-type litters or stretchers (figure 4–25). The litters are installed in two rows, one down each side of the cabin. Each has three groups of litters stacked four deep. The bottom litter of each group rests on the cabin floor and the three litters above it are clamped by their outboard poles to the cabin side panels. The inboard poles of the litters are supported by straps running from fittings in the ceiling to the cargo tie-down rings in the floor. When all litters are installed, the forward half of the cargo door is blocked. Twenty-eight litter straps are stowed in pockets along the cabin side panels. Fourteen straps are equipped with four clamps each to support the inboard litter poles, and the outboard end of the aft poles in the way of the cargo door and the escape hatch. Ten short straps, with one clamp each, are used to hold the bottom outboard litter poles to the floor. Four straps, with pockets sewed into one end, are looped over the forward outboard handles of the two forward upper litters on each side to prevent the litters from sliding forward, and are clipped to rings at the forward window frames. To

Figure 4—27. Main Rotor Blade Folding and Tail Rotor Coning Control Panel

actuate the clamps, both those attached to the cabin panels and those supported by the straps, insert the litter pole in the jaw and pull the short strap to tighten the clamps. The strap is locked by a cam which may be released to open the clamp.

LITTER LOADING PROCEDURE.

1. Attach inboard straps to overhead.
2. Load patient feet first, feet forward.
3. Load the front of the helicopter first, a group of four on the left side, then a group on the right side, etc.
4. Load litters from top to bottom.
5. Attach litter to cabin bulkhead clamps first.
6. Secure litter to inboard litter straps.
7. Secure bottom litter outboard pole to floor.
8. Secure inboard litter straps to cabin floor after the last litter in that group is loaded.
9. Strap patient to litter (this may be accomplished prior to loading).

MAIN ROTOR BLADE AND PYLON FOLDING.

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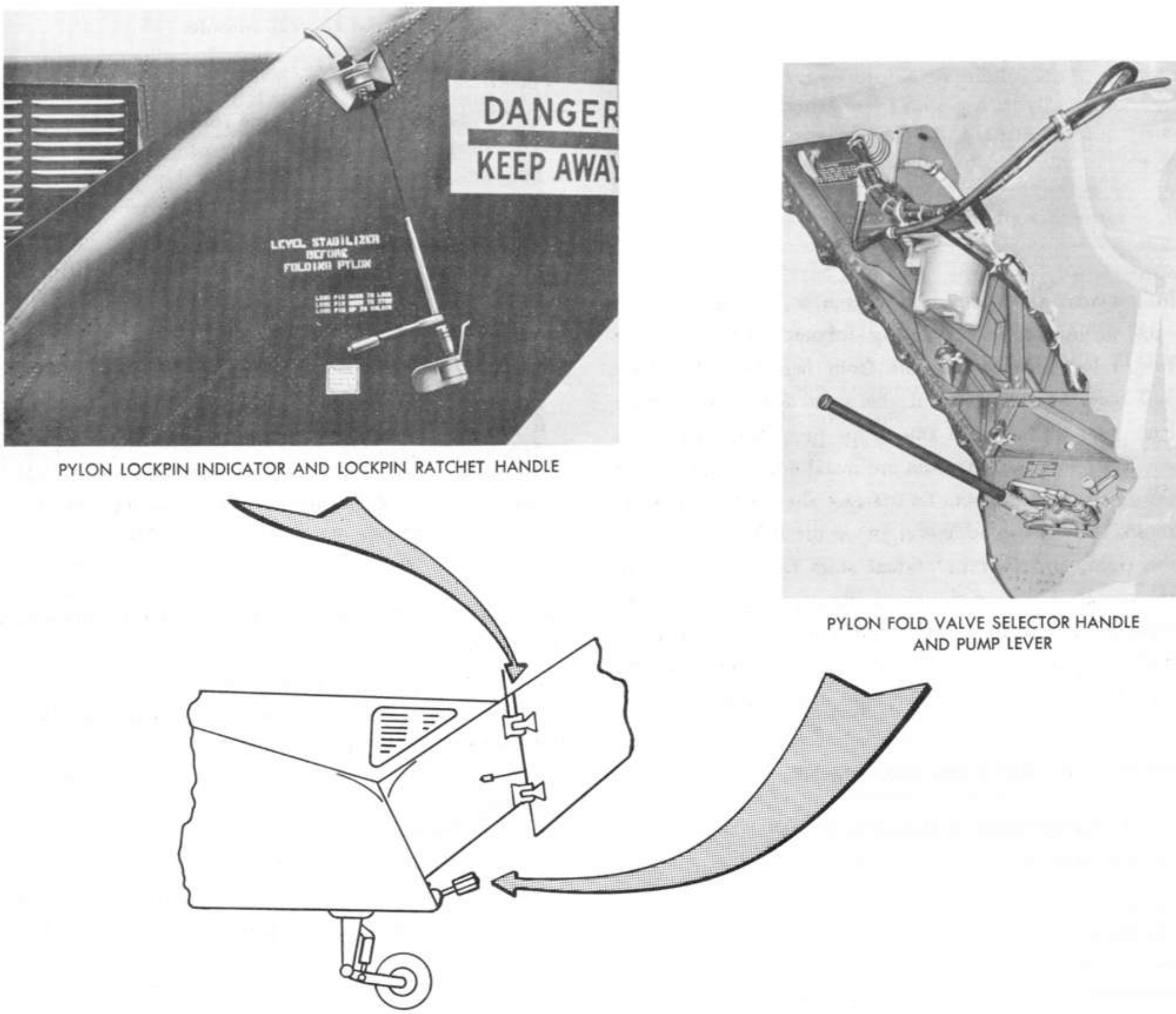


PYLON LOCKPIN INDICATOR AND LOCKPIN RATCHET HANDLE

PYLON FOLD VALVE SELECTOR HANDLE AND PUMP LEVER

Figure 4–29. Pylon Fold Controls (BuNos 140314 and Subsequent)

FULLY-AUTOMATIC MAIN ROTOR BLADE FOLDING.

On BuNos 140314 and subsequent, the main rotor blades may be folded back over the fuselage by a fully automatic operation which is controlled in the cockpit by either pilot. The operation may be halted at any stage of the folding or unfolding cycle and reversed, if so desired. Hydraulic power from the utility hydraulic system is used in conjunction with hydraulic and electrical sequencing. The folding is accomplished by actuating the proper switches in sequence on the control panel located on the control console and observing the blade positioning and folding in conjunction with the appropriate lights on the panel. The left engine should be running at a minimum of 1500 rpm for blade folding. If the right engine is running, the second stage tandem servo pressure will drop to zero when the master folding switch is turned on. The automatic stabilization servo system is turned on through the automatic stabilization servo switch to provide enough power to the cyclic control stick to operate the controls slightly during control lockpin engagement. The No. 1 blade positions directly aft and does not fold. When the blade folding cycle is complete, the blades are stowed by ground crew in appropriate racks attached to the fuselage.

MANUAL ROTOR PARKING LOCK.

A manual rotor parking lock is provided, on BuNos 140314 and subsequent, to assure that the main rotor will not rotate when the blades are in the folded position. When the lock is engaged, the flight position indicator light is off, there is no electrical power to close the safety valve, and a microswitch interrupts the clutch electrical circuit, preventing clutch engagement when the mechanical lock is engaged. The manual rotor parking lock handle has a hinged cover with a decal warning, and is located overhead in the cabin to the right of the cargo hoist monorail and forward of the cargo door. To engage the mechanical lock, pull the manual rotor parking lock handle and turn it to engage in the handle support. This rotates the locking pin into engagement with the main rotor brake disc. To disengage the mechanical lock, remove the manual rotor parking lock handle from the handle support and allow the handle to return to the disengage position. Tension springs release the locking pin from engagement with the main rotor brake disc, and the microswitch closes the circuit, allowing engagement of the clutch.

WARNING

Do not use the manual rotor parking lock unless the main rotor blades are in the folded position.

BLADE FOLDING CONTROL PANEL (AUTOMATIC).

The blade folding control panel (figure 4–27) is marked MAIN BLADE FOLD TAIL ROTOR CONING and is located on the control console.

SAFETY VALVE SWITCH. A guarded switch, marked SAFETY VALVE, with two positions OPEN and CLOSED, is located on the blade folding control panel. A red warning light to the right of the safety valve switch indicates the OPEN position. The switch operates a motor-driven three-way selector valve located on the bulkhead on the left side of the cabin adjacent to the gear box, and prevents inadvertent application of hydraulic pressure to the rotor head and automatic rotor brake during flight. When this switch is in the OPEN position, the flight position indicator light will go off.

BLADE FOLDING MASTER SWITCH.

The switch, marked MASTER, located on the blade folding control panel, completes the circuit to furnish electrical power to actuate the hydraulic sequences for automatic blade folding. The fold power warning light, marked FOLD PWR, indicates that folding power is available. This light will come on when the master and safety valve switches are on, and the hand rotor brake is released.

BLADE FOLD-UNFOLD SWITCH.

A three position blade fold-unfold switch, marked BLADES FOLD, UNFOLD, and OFF, located on the blade folding control panel, is used to select the folding cycle, which consists of rotor head positioning and blade folding operations, or the unfolding cycle.

BLADES UNFOLDED INDICATOR LIGHT.

An amber light, marked BLADES UNFOLDED, located on the blade folding control panel, will come on when all the blades are in their unfolded position. This light will go off during the folding cycle when any control lockpin advances and the control lockpins advanced light comes on. At the end of the unfolding cycle, the blades unfolded indicator light will come on when the control lockpins advanced light goes off and all pins are fully disengaged.

FLIGHT POSITION INDICATOR LIGHT.

A green light, marked FLIGHT POS., located on the blade folding control panel, will come on when the blades are completely unfolded, the tail rotor blades are not coned, the pylon hinge pins are properly positioned, and the safety valve switch is closed. When the safety valve switch is open, the flight position indicator light will go off.

NO. 1 BLADE POSITION INDICATOR LIGHT.

An amber light, marked NO. 1 BLADE POS., located on the blade folding control panel, will come on during the folding cycle after the rotor head positions with the No. 1 blade directly aft.

CONTROL LOCKPINS ADVANCED INDICATOR LIGHT.

An amber light, marked CONT. LOCK PINS ADV., located on the control panel, will come on during the folding cycle as soon as one flight control lockpin moves forward and will remain on until all the flight control lockpins are fully disengaged at the end of the unfolding cycle.

BLADES FOLDED INDICATOR LIGHT.

An amber light, marked BLADES FOLDED, located on the blade folding control panel, will come on when all blades are folded. During the unfolding cycle, this light will go off when any blade moves forward.

AUTOMATIC BLADE FOLDING PROCEDURE.

1. Droop and anti-coning stops — IN PLACE.
2. Quadrant throttles — CLOSED AND LOCKED. Left engine — 1500 rpm minimum.
3. ASE servo switch — Check ON.
4. Automatic stabilization equipment — Standby (Check pitch and roll modes for center).
5. Collective pitch control — 10 DEGREES ABOVE LOW PITCH AND LOCK WITH PIP PIN.

Note

If narrow chord blades (14° 21.5-inch chord) are installed: Collective pitch control — MINIMUM PITCH AND LOCKED.

6. Cyclic control stick — NEUTRAL. (Locked with stick positioner.)
7. Stick trim release switch — DEPRESS to center trim system.
8. Rotor brake lever — ON.
   Rotor brake warning light on. Ascertain position of No. 1 blade.
9. Safety valve switch — OPEN. Safety valve open warning light on, flight position indicator light off.
10. Blade folding master switch — ON.
11. Rotor brake lever — OFF.
    Fold power on warning light on. Rotor brake warning light on.

Note

If the right engine is operating, the second stage servo hydraulic low pressure warning light should light and the second stage servo hydraulic pressure gage should indicate zero.

12. Blade fold-unfold switch — FOLD.
    Observe the following sequence:
    a. No. 1 blade positions aft.
       No. 1 blade position indicator light on.

Note

Check visually that the two forward blades are positioned an equal distance from the center line of the helicopter. If the blades are not in the correct position, refer to EMERGENCY FOLDING PROCEDURE in this section.

    b. Dampers position all blades against their autorotation stops.
    c. Control lockpins advance indicator light on. Blades unfolded warning light off as soon as one flight control lockpin advances.

Note

It may be necessary to actuate the cyclic control stick slightly to seat all the lockpins.

    d. Blades folded indicator light on when all blades are folded.
13. Notify ground personnel to engage manual rotor parking lock.
14. Bicycle clips — INSTALL.
    Crewman installs clips on all five dampers to prevent blades from drifting when engines are shut down.
15. Blade fold-unfold switch — OFF.
    Leave safety valve open.
16. Stow blades in racks.
17. Rotor brake lever — ON.
    a. Fold power on warning light off.
    b. No. 1 blade position indicator light off.
18. Blade folding master switch — OFF.
    a. Blades folded indicator light will remain on.
19. Secure engines.

AUTOMATIC BLADE UNFOLDING PROCEDURE.

1. Before starting the engine, apply electrical power and check the following:
   a. Safety valve switch — OPEN, ASE servo switch — ON, master switch — ON, blades fold-unfold switch — OFF, push ASE disengage button on cyclic control stick, signal crewman to disengage manual rotor parking brake, rotor brake lever — ON, collective pitch control — 10 DEGREES ABOVE LOW PITCH AND LOCK WITH PIP PIN, and cyclic control stick — NEUTRAL (locked with stick positioner), No. 1 blade — AFT.

Note

If narrow chord blades (14° 21.5-inch chord) are installed: Collective pitch control — MINIMUM PITCH AND LOCKED.

   b. Safety valve open warning light on, blades folded indicator light on, and control lockpin advanced indicator light on.

c. Storage rack — Blades free.

2. Quadrant throttles — CLOSED AND LOCKED. Left engine 1500 rpm minimum. Check utility hydraulic pressure.
3. Rotor brake lever — OFF. Fold power on warning light on.
4. Blade fold-unfold switch — UNFOLD. Observe the following sequence:
   a. Blades folded indicator light off when first blade moves.
   b. Control lockpins advanced. Indicator light off when all pins are fully disengaged.
   c. Blades unfolded indicator light on.
5. Crewman removes "bicycle clips" from dampers.
6. Disengage pip-pin lock on collective pitch control.
7. Blade folding master switch — OFF. Fold power on warning light off.
8. Blade fold-unfold switch — OFF.
9. Safety valve switch — CLOSED. Safety valve open warning light off. Flight position indicator light on provided the tail rotor blades are in the flight position.
10. After clutch engagement and transmission oil pressure is indicated, observe the following sequence:
    a. Blades unfolded indicator light off.
    b. Flight position indicator light off.
    c. Transmission low pressure warning light off.

EMERGENCY FOLDING PROCEDURE.

1. If improper blade positioning is experienced during automatic blade folding and it is necessary to complete blade folding rapidly, proceed as follows:
   a. Blade folding master switch — OFF.
   b. Rotor brake lever — OFF.
   c. No. 1 blade — AFT (position manually).
   d. Blade folding master switch — ON (proceed with automatic operation).
2. In case of electrical power failure to the blade folding system, except that power is available to the safety valve switch, proceed as follows:
   a. Rotor brake lever — OFF.
   b. No. 1 blade — DIRECTLY AFT (position manually or with clutch drag).
   c. Rotor brake lever — ON. Engage manual rotor parking lock.
   d. Collective pitch control — 10 DEGREES ABOVE LOW PITCH AND LOCK WITH PIP PIN.

Note

If narrow chord blades (14° 21.5-inch chord) are installed: Collective pitch control — MINIMUM PITCH AND LOCKED.

   e. Cyclic control stick — NEUTRAL AND LOCKED (have ground crew check control lockpin alignment).
   f. Safety valve switch — OPEN.
   g. Trip "Fold" manual override, and observe the following sequence:
      (1) Dampers position.
      (2) Control locks engage.
      (3) Blade lockpins retract.
      (4) Blades fold.
3. In case of complete electrical failure or hydraulic failure, proceed as follows:
   a. Rotor brake lever — OFF.
   b. No. 1 blade — DIRECTLY AFT (position manually).
   c. Rotor brake lever — ON. Engage manual rotor parking lock.
   d. Rotate blades about feathering axis to align control lockpins with holes.

Note

A crew of four men is recommended to expedite manual blade folding.

   e. Control lockpins — ENGAGE (manually by actuating pinion at aft end of control locks).
   f. Blade lockpins — DISENGAGE (manually by cutting safety wire and threading nuts down on the pin).
   g. Remove cone bolt from the gear box arm.

Note

To avoid the possibility of damaging the cone bolt and to facilitate the subsequent unfolding operation, the following procedure should be used when removing the cone bolt:

1. Disconnect hydraulic lines to hydraulic motors.
2. Remove nut from bottom of cone bolt.
3. Remove hydraulic motor and rotate gear box input shaft until load on the cone bolt is released.
4. Remove cone bolt by turning head counterclockwise.

   h. Fold blades manually with blade crutches and stow them in the blade racks.

EMERGENCY UNFOLDING PROCEDURE.

Manual unfolding of the blades is accomplished in reversed order from folding, but it is recommended that hydraulic and electrical power be available prior to the next flight which would make manual unfolding unnecessary.

PYLON FOLDING.

On BuNos 140314 and subsequent, the tail rotor pylon may be folded by use of a hydraulic pump located in the tail cone extension and accessible to a crew member stationed outside the aircraft. Two-position folding is provided for light maintenance above deck; and heavy maintenance and/or stowage below deck. The tail rotor pylon folds forward and to the right for a total of 174 degrees about its folding hinge axis. Folding or unfolding in the 0 to 115 degree range is directionally controlled by the manually operated selector valve; folding or unfolding in the 115 to 174 degree range requires, in addition, the manual actuation of the sequence valve.

HAND PUMP.

A standard 3000-psi hand pump, located in the tail cone extension and accessible through a removable fairing, is used to furnish hydraulic pressure to fold and unfold the tail rotor pylon.

SELECTOR VALVE.

A manually-operated selector valve, marked FOLD, OFF, UNFOLD, is located in the tail cone extension adjacent to the hand pump. The selector valve is used to select the folding or unfolding direction.

SEQUENCE VALVE.

A manually-operated sequence valve, with built in thermal relief, is located at the folding link of the pylon. This valve is actuated by a T-handle Teleflex control which is located in the tail cone extension above the selector valve.

TAIL ROTOR CONING SWITCH.

The tail rotor coning switch, a guarded switch, marked CONED and FLT. POS., is located on the main rotor blade folding and tail rotor coning control panel (figure 4–27). When the switch is placed in the CONED position, the tail rotor blades will cone and the blades coned red warning light, located on the control panel, will come on. When the switch is placed in the FLT. POS., the tail rotor blades uncone and the warning light will go off when unconing is accomplished.

PYLON LOCKPIN RATCHET HANDLE AND INDICATOR.

The pylon lockpins, located on the left side of the fuselage at the base of the pylon, are pulled by actuating a ratchet handle (figure 4–29) fore-and-aft. The lower pin is extracted upward and the upper pin downward. The ratchet action of the handle is reversed for seating the lockpins after the pylon is unfolded by twisting the end of the handle 180 degrees. When not in use, the ratchet handle is pushed forward where it is retained by a spring-loaded latch. When the pylon lockpins retract, a red pylon lockpin indicator (figure 4–29) appears from a slot in the fuselage skin above the lockpins. As the pins unlock, an amber pylon pins unlocked indicator light, located on the control panel, will come on. When the pylon hinge pins are locked, the pylon pins unlocked indicator light will go off.

NORMAL OPERATION. (Main rotor blades folded.)

To fold the pylon:

1. Tail rotor coning switch – CONED. (Blades coned warning light on.)
2. Pylon hinge pins – RETRACT. (Pylon pins unlocked indicator light on.)
3. Selector valve – FOLD.
4. Actuate hand pump until pylon begins to fold by its own weight.
5. Actuate selector valve to control folding to 115 degrees.

Note

If the helicopter is to be serviced in this position (115 degrees), do not leave the pylon folded longer than 30 minutes without installing jury strut.

6. Manually unfold and hold the No. 3 main rotor blade clear of the tail rotor.

Note

Because the blade folding gear box is irreversible, it is necessary to remove the cone bolt from the gear box arm and manually hold the No. 3 main rotor blade out of the way of the pylon folding track.

7. Sequence valve – Pull Teleflex control (control pylon fold to 174 degrees).
8. Install jury strut.
9. Return No. 3 main rotor blade to folded position.
10. Install cone bolt to gear box arm.
11. Selector valve – FOLD.

To unfold the pylon:

1. Remove cone bolt from gear box.
2. Manually unfold and hold No. 3 main rotor blade clear of tail rotor.
3. Remove jury strut.
4. Selector valve – UNFOLD.
5. Sequence valve – Pull Teleflex control. (Hold while actuating hand pump until pylon passes 115-degree position.)
6. Actuate hand pump until pylon is fully unfolded.

7. Selector valve – FOLD.
8. Pylon hinge pins – LOCK. (Pylon pins unlocked indicator light off.)
9. Return No. 3 main rotor blade to folded position.
10. Tail rotor coning switch – FLT. POS. (Blades coned warning light off.)
11. Install cone bolt to gear box arm.

MISCELLANEOUS EQUIPMENT.

CHECK LIST.

A check list is located in the center of the instrument panel. It is lighted by actuating the console and panel light rheostat on the overhead switch panel.

ANTI-GLARE PANEL.

The anti-glare panel consists of a one-piece nylon curtain used when it is desired to shield the pilot's compartment from glare originating in the cabin. An aluminum alloy tube, which is fastened by two tabs to the curtain, engages two spring clips to secure the forward end of the curtain to the flight deck. The rear end of the curtain is held by snap fasteners at the forward end of the radio equipment deck. When not in use, the anti-glare panel is rolled and fastened by snap fasteners at the radio deck.

BLIND-FLYING PANELS.

A set of semi-opaque orange panels may be attached to the inside of the pilot's compartment canopy to cover the windows for blind-flying practice.

GUN PORTS.

Each nose door is equipped with a gun port facing forward that is covered by a small door. Each door hinges inward at the bottom and is held closed by two fasteners at the top.

MOORING FITTINGS.

Five mooring fittings are provided on the helicopter. Each main landing gear strut has a fitting (22, figure 1–2) located on the inboard and outboard side. A fitting (30, figure 1–2) is located at the rear of the fuselage aft of the tail wheel.

UTILITY RECEPTACLES.

Four capped electrical utility receptacles are connected to the secondary bus. Two receptacles are located in the pilot's compartment, one on each side panel below the sliding windows. Two receptacles are located on the right cabin side panel near the interphone stations, one forward beneath the radio deck and one aft near the cargo door.

CANTEENS.

A 1-quart canteen (3, figure 3–4) is mounted on the left side of the aft cabin bulkhead. A second canteen (7) is mounted on the right side panel of the cabin over the middle window.

WINDSHIELD WIPERS.

Hydraulically operated windshield wipers are installed in front of the pilot and copilot. The wipers are controlled by a knob, marked ON and OFF, located just left of center below the bottom edge of the instrument panel. Through the hydraulic control they are synchronized, speed controlled, and have a bypass valve for continued operation in case of malfunction of one of the units. Hydraulic power is obtained through the wheel brake system from the utility hydraulic system.

MAP CASE AND CHARTBOARD.

A map case is located on the radio deck over the pilot's compartment access ladder. A MK-6A chartboard is stowed underneath the map case.

BALANCE COMPUTER.

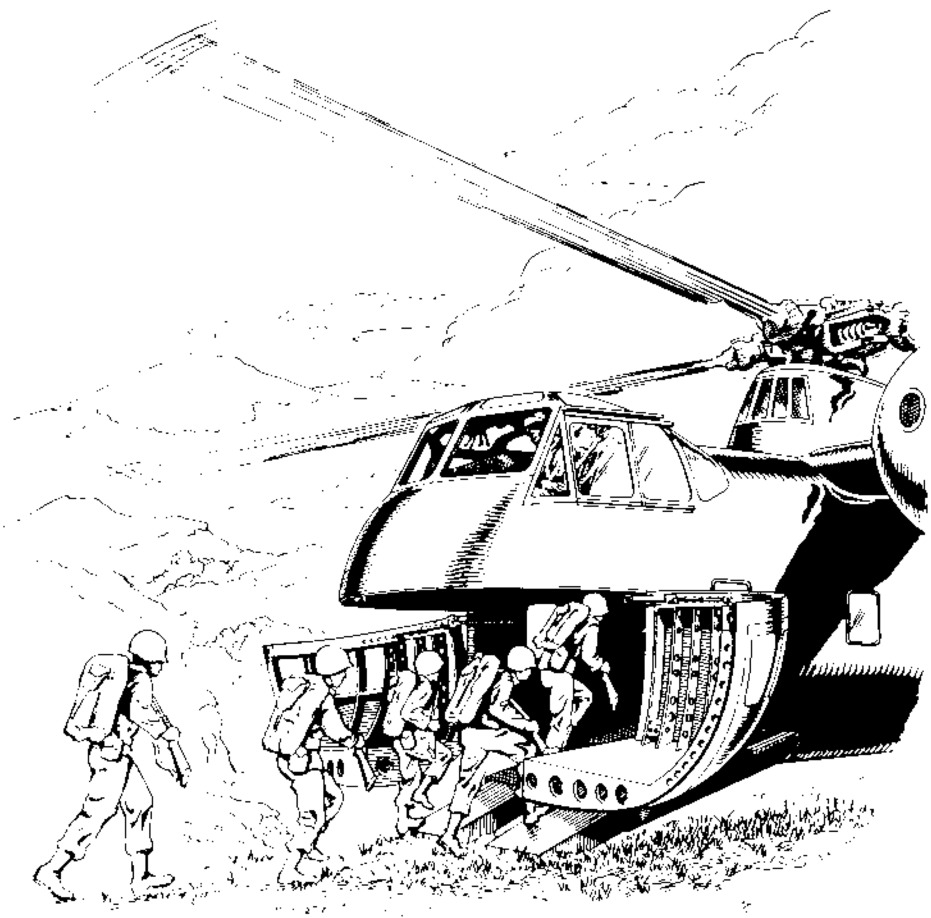
A slide-rule-type cabin loading balance computer is stowed in a case mounted on the copilot's inboard seat support.

RELIEF TUBES.

Relief tubes are installed in brackets under the pilot's and the copilot's seats. A third relief tube is installed in the aft right corner of the cabin.

DYNAMIC VIBRATION ABSORBERS.

On BuNos 145866 and subsequent, and helicopters modified in accordance with Aircraft Service Change No. 35A, a dynamic vibration absorber is installed in the fuselage above the left and right nose doors to reduce vibration level. For best operating results, the smooth frequency bands of the dynamic vibration absorbers should be tuned to 2600 engine rpm.



SECTION V
OPERATING LIMITATIONS

GENERAL.

Operating limitations are derived from flight testing and flight research. These limitations insure your safety and help to obtain maximum utility from the helicopter and its equipment. The instruments are marked, as shown in figure 5–1, to serve as a constant reminder of airspeed and engine limitations; however, additional limitations on operational procedures, maneuvers, and loading are given in the following paragraphs.

MINIMUM CREW REQUIREMENTS.

The minimum crew required to operate the helicopter under normal nontactical conditions is a pilot and a copilot. Additional crew members, as required, will be added at the discretion of the Commanding Officer.

ENGINE LIMITATIONS.

POWER LIMITATIONS.

TWO-ENGINE OPERATION. Take-off power, 2100 bhp per engine, is limited to 5 minutes operation. Power settings for take-off power at sea level are 2700 rpm and 57.5 inches Hg with RICH mixture.

Normal rated power is 1725 bhp per engine. Power settings for normal rated power at sea level are 2600 rpm and 46.0 inches Hg with NORMAL mixture.

SINGLE-ENGINE OPERATION. Take-off power, 2100 bhp, is limited to 5 minutes operation. Power settings

FUEL GRADE 115/145

ENGINE TACHOMETER

14° 21.5 INCH CHORD BLADES (S1515-20400)

- (red) 2000 RPM—MINIMUM FLIGHT
- (green) 2200 TO 2600 RPM—NORMAL
- (red) 2700 RPM—MAXIMUM (5 MINUTES)

APPARENT ROTOR RPM

- (tan) 1960 RPM—MINIMUM FLIGHT
- (green) 2200 TO 2600 RPM—NORMAL
- (red) 3020 RPM—MAXIMUM

MANIFOLD PRESSURE

- (green) 18 to 46.0 IN. HG—NORMAL
- (red) 57.5 IN. HG—MAXIMUM (5 MINUTES)

ENGINE TACHOMETER

8° 23.65 INCH CHORD BLADES (S1515-20701)

- (red) 2200 RPM—MINIMUM FLIGHT
- (green) 2400 TO 2600 RPM--NORMAL
- (red) 2700 RPM—MAXIMUM (5 MINUTES)

APPARENT ROTOR RPM

- (tan) 1960 RPM—MINIMUM FLIGHT
- (green) 2200 TO 2600 RPM—NORMAL
- (red) 3020 RPM—MAXIMUM

FIRST STAGE HYDRAULIC PRESSURE

- (red) 2600 PSI—MINIMUM
- (green) 2600 TO 3150 PSI—NORMAL
- (red) 3150 PSI—MAXIMUM

TRANSMISSION OIL TEMPERATURE (MAIN GEAR BOX)

- (red) −15°C—MINIMUM
- (green) 40 TO 120°C—NORMAL
- (red) 140°C—MAXIMUM

TRANSMISSION OIL PRESSURE (MAIN GEAR BOX)

- (red) 15 PSI—MINIMUM
- (green) 25 TO 60 PSI—NORMAL
- (red) 120 PSI—MAXIMUM

UTILITY SYSTEM HYDRAULIC PRESSURE

- (red) 2600 PSI—MINIMUM
- (green) 2600 TO 3150 PS—NORMAL
- (red) 3150 PSI—MAXIMUM

SECOND STAGE SERVO HYDRAULIC PRESSURE

- (red) 1750 PSI—MINIMUM
- (green) 1750 TO 2150 PSI—NORMAL
- (red) 2150 PSI—MAXIMUM

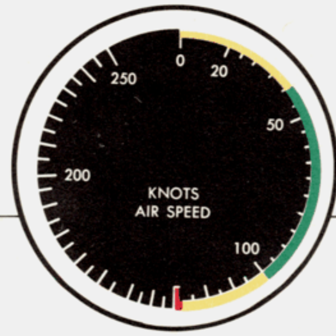
EMERGENCY SYSTEM HYDRAULIC PRESSURE

- (red) 1500 PSI—MAXIMUM

Figure No. 5–1. Instrument Range Markings (Sheet 1)

AIRSPEED INDICATOR
14° 21.5 INCH CHORD BLADES (S1515-20400)

- (Yellow) 0 TO 40 KNOTS—CAUTION LOW ALTITUDE
- (Green) 40 TO 90 KNOTS—NORMAL
- (Yellow) 90 TO 110 KNOTS—POSSIBLE BLADE STALL
- (Red) 110 KNOTS—MAXIMUM

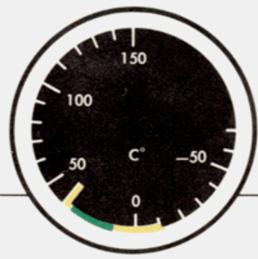


AIRSPEED INDICATOR
8° 23.65 INCH CHORD BLADES (S1515-20701)

- (Yellow) 0 TO 40 KNOTS—CAUTION LOW ALTITUDE
- (Green) 40 TO 100 KNOTS—NORMAL
- (Yellow) 100 TO 130 KNOTS—POSSIBLE BLADE STALL
- (Red) 130 KNOTS—MAXIMUM

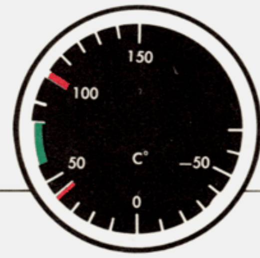
CYLINDER HEAD TEMPERATURE

- (Red) 100°C—MINIMUM
- (Green) 150 TO 200°C—NORMAL
- (Red) 260°C—MAXIMUM



CARBURETOR AIR TEMPERATURE

- (Yellow) —10 TO + 15°C—CAUTION ICING
- (Green) 15 TO 38°C—NORMAL
- (Yellow) 38°C—MAXIMUM WITH CARBURETOR HEAT



ENGINE OIL TEMPERATURE

- (Red) 40°C—MINIMUM
- (Green) 60 TO 80°C—NORMAL
- (Red) 100°C—MAXIMUM

FUEL PRESSURE

- (Red) 14 PSI—MINIMUM
- (Green) 21 TO 23 PSI—NORMAL
- (Red) 25 PSI—MAXIMUM

ENGINE OIL PRESSURE

- (Red) 40 PSI—MINIMUM
- (Green) 80 TO 90 PSI—NORMAL
- (Red) 100 PSI—MAXIMUM

Figure No. 5—1. Instrument Range Markings (Sheet 2)

MAXIMUM MANIFOLD PRESSURES AT VARIOUS ALTITUDES UNDER STANDARD ATMOSPHERIC CONDITIONS

SINGLE-ENGINE OPERATION

POWER	MIXTURE	ENGINE RPM	SEA LEVEL	ALTITUDE— FEET					
				2000	4000	6000	8000	10,000	12,000
TAKE-OFF	RICH	2700	57.5	56.2	F.T.	F.T.	F.T.	F.T.	F.T.
NORMAL RATED	NORMAL	2600	50.5	49.8	49.0	F.T.	F.T.	F.T.	F.T.
	NORMAL	2500	48.0	47.5	47.0	F.T.	F.T.	F.T.	F.T.

ENGINE: R-2800-54 BASED ON: FLIGHT TEST
FUEL GRADE: 115/145 DATA AS OF: DECEMBER 1, 1958

(1) Take-off (2100 BHP) limited to 5 minutes of operation.
(2) F.T. — full throttle.
(3) All manifold pressures are in inches of mercury.

Figure 5—2. Maximum Manifold Pressures — Single-Engine Operation

for take-off power at sea level are 2700 rpm and 57.5 inches Hg with RICH mixture.

Normal rated power is 1900 bhp. Power settings for normal rated power at sea level are 2600 rpm and 50.5 inches Hg with NORMAL mixture.

ENGINE OVERSPEED LIMITS.

2800 rpm to 3100 rpm — Allowable rpm.

3100 rpm to 3350 rpm — Engine and fan assembly inspetcion required. For engine inspection, refer to GREB No. 167. Check fan assembly for specified clearance between fan blade tips and contravane assembly.

3350 rpm and over — Remove the engine and fan assembly for overhaul.

MANIFOLD PRESSURE.

The manifold pressure gage range markings, shown in figure 5—1, are for sea level conditions. As altitude is increased, up to the critical altitude, manifold pressure must be reduced in accordance with figures 5—2 and 5—3 to avoid exceeding the power limitations as set by the engine manufacturer. Above critical altitude, full throttle may be used.

CARBURETOR AIR TEMPERATURE.

The cautionary range marking (yellow arc), located at

MAXIMUM MANIFOLD PRESSURES AT VARIOUS ALTITUDES UNDER STANDARD ATMOSPHERIC CONDITIONS

TWO-ENGINE OPERATION

POWER	MIXTURE	ENGINE RPM	SEA LEVEL	ALTITUDE— FEET					
				2000	4000	6000	8000	10,000	12,000
TAKE-OFF	RICH	2700	57.5	56.2	F.T.	F.T.	F.T.	F.T.	F.T.
NORMAL RATED	NORMAL	2600	46.0	45.1	44.4	43.7	43.0	F.T.	F.T.
	NORMAL	2500	46.0	45.2	44.6	44.0	F.T.	F.T.	F.T.

ENGINE: R-2800-54 BASED ON: FLIGHT TEST
FUEL GRADE: 115/145 DATA AS OF: DECEMBER 1, 1958

(1) Take-off (4200 BHP) limited to 5 minutes of operation.
(2) F.T. — full throttle.
(3) All manifold pressures are in inches of mercury.

Figure 5—3. Maximum Manifold Pressures — Two-Engine Operation the low end of the carburetor air temperature indicator scale (figure 5—1), is of assistance in the use of carburetor heat in order to prevent carburetor icing. Whenever carburetor air temperature is within the cautionary range, heated air should be applied. When using heated air, the carburetor air lever should be adjusted to hold carburetor air temperature above the lower cautionary range and below 38°C.

Note

Reduce maximum allowable manifold pressure 1 inch Hg for each 6°C the carburetor air temperature exceeds 38°C.

TRANSMISSION LIMITATIONS.

Operating limitations of the transmission system are governed by the main gear box oil pressure and temperature.

GROUND OPERATION – CLUTCHES DISENGAGED.

Prolonged ground operation, with the clutches disengaged, is not recommended; however, when necessary, ground operation should be accomplished at between 1500 and 1600 rpm to minimize the possibility of clutch overheating. Overheating of the clutches can damage the oil seals causing leakage of oil and hydraulic fluid and failure of the clutches. When ground operation, with the clutches disengaged, is necessary, the duration of ground operation should be varied in accordance with ambient air temperature as shown in figure 5—4 and engine speed should be between 1500 and 1600 rpm.

MAIN ROTOR LIMITATIONS.

Rotor speed should never be allowed to drop below 1960 apparent rpm (yellow radial line) during autorotation as the inertia of the rotating blades at this speed is not sufficient to accomplish a safe power-off landing. Apparent rotor rpm should never exceed 3020 rpm (red radial line) to assure a proper margin below the critical speed of the tail rotor drive shaft. Note that true rotor rpm cannot be read directly from the tachometer. The maximum recommended wind velocity for rotor clutch engagement and disengagement is 60 knots with the helicopter headed into the wind.

CAUTION

On helicopters prior to BuNo 145875 unless modified by ASC No. 44A, to prevent clutch failure and resulting engine overspeed, the maximum rate of collective stick motion from low to high pitch shall not exceed that corresponding to full travel in two seconds.

TRANSMISSION AND ROTOR OVERSPEED.

Overspeeding of the rotor system imposes severe loads on the rotor head and the transmission system. Whenever the rotor rpm exceeds the red radial line on the rotor scale of the tachometer, the maximum overspeed and duration of the overspeed condition should be noted on the appropriate form. The following corrective action should be taken in the event of a main rotor overspeed:

MAXIMUM DURATION — GROUND RUN CLUTCHES DISENGAGED		
AMBIENT AIR TEMPERATURE		DURATION MINUTES
°C	°F	
−18	0	60
−4	25	45
10	50	30
24	75	20
38	100	10

Figure 5—4. Maximum Duration — Ground Run Clutches Disengaged

Note

Components removed for overspeed should be identified with a tag or marking. The log card of the component should state the reason for removal and the actual rpm attained during overspeed. Any component that does not have an overspeed rpm entered on its log card must be considered to have exceeded 3640 rpm.

3150 to 3360 apparent rpm — Inspect following components:

1. Main rotor blades for dents, cracks, buckeld pocket skin, and bonded joints for separation beyond specified limits. Tip block rivets for security, and tip caps for distortion and loose screws.
2. Tail rotor blades for dents, cracks, distortion, and separation of bonding of spar to trailing edge. Remove tip caps and check for security of balance weights and distortion of points of attachment of weights.

Note

Any main or tail rotor blade with distorted rivets at the tip end should be returned to the contractor for replacement of rivets.

3. Main and tail rotor head assemblies for damage or distortion indicating yield of material. Control linkages for damage or distortion.
4. Tail rotor drive shaft brackets for cracks extending from attachment holes. Drive shaft rubber couplings for bond failure.

Figure 5–5 deleted.

5. Magnetic plugs and strainers in gear boxes for foreign material.

3360 to 3640 apparent rpm — Inspect or remove the following components:

1. Inspect main and tail rotor blades and control linkages as in steps 1., 2., and 3. of 3150 to 3360 apparent rpm overspeed.
2. Remove main rotor head assembly and tail rotor assembly for overhaul.
3. Remove and inspect main drive shafts and flanges for cracks (fluorescent dye penetrant or Zyglo) and straightness (run-out). Inspect all flanges for elongated bolt holes. Inspect rubber couplings for bond separations and rubber coupling jaws for cracks (Zyglo).
4. Remove all sections of tail drive shaft and inspect drive shafts, flanges, and rubber coupling jaws for cracks (fluorescent dye penetrant or Zyglo) and straightness (run-out). Inspect tail drive shaft brackets for cracks extending from attachment holes. Inspect rubber couplings for bond separation beyond specified limits.
5. Inspect airframe for cracks in area of rotor brake support.

MAXIMUM LEVEL FLIGHT AIRSPEEDS — 72-FOOT MAIN ROTOR
AIRSPEEDS LIMITED BY NORMAL POWER OR BLADE STALL
TWO-ENGINE OPERATION
CONFIGURATION: CLEAN, GEAR UP

RPM	ALTITUDE (FEET) / GROSS WEIGHT (LBS)	SEA LEVEL	2000	4000	6000	8000	10,000
		INDICATED AIRSPEED — KNOTS					
2500	23,500	130	124	117*	107*	96*	83*
	26,000	125	117*	107*	95*	80*	60*
	28,500	118	108*	95*	79*	52*	—
	31,000	108*	94*	77*	44*	—	—
2600	23,500	130	128	123	115*	104*	91*
	26,000	127	122	115*	103*	88*	68*
	28,500	120	115	103*	87*	60*	—
	31,000	114	102*	85*	52*	—	—

*Denotes blade stall.
ENGINES: (2) R-2800-54 BASED ON: FLIGHT TEST
FUEL GRADE: 115/145 DATA AS OF: DECEMBER 1, 1958

Figure 5–6. Maximum Level Flight Airspeeds — 72-Foot Main Rotor

MAXIMUM LEVEL FLIGHT AIRSPEEDS — 72-FOOT MAIN ROTOR
AIRSPEEDS LIMITED BY NORMAL POWER OR BLADE STALL
TWO-ENGINE OPERATION
CONFIGURATION: ONE 150 OR 300 GAL. AUXILIARY TANKS, GEAR UP

RPM	ALTITUDE (FEET) / GROSS WEIGHT (LBS)	SEA LEVEL	2000	4000	6000	8000	10,000
		INDICATED AIRSPEED — KNOTS					
2500	23,500	122	115	114*	104*	94*	81*
	26,000	120	113	105*	94*	78*	58*
	28,500	114	106*	94*	78*	49*	—
	31,000	104*	93*	76*	44*	—	—
2600	23,500	123	121	118	112*	102*	89*
	26,000	120	117	112	102*	86*	66*
	28,500	117	111	102*	86*	57*	—
	31,000	112	101*	84*	52*	—	—

*Denotes blade stall.
ENGINES: (2) R-2800-54 BASED ON: FLIGHT TEST
FUEL GRADE: 115/145 DATA AS OF: DECEMBER 1, 1958

Figure 5–7. Maximum Level Flight Airspeeds — 72-Foot Main Rotor

6. Inspect magnetic plugs and strainers in gear boxes for foreign material.
7. Inspect tail drive shaft disconnect for abnormal wear of teeth.

Over 3640 apparent rpm — Inspect or remove the following components:

1. Remove main and tail rotor blades. Condemn and salvage locally.
2. Remove tail rotor assembly. Condemn and salvage locally.
3. Visually inspect main and tail rotor control linkages for damage and distortion.
4. Remove main rotor head assembly. Return star assembly dampers, rotating and stationary scissors assemblies, and torque tube assemblies for overhaul. Condemn main rotor hub assembly and salvage locally.
5. Inspect tail rotor drive shaft brackets for alignment and tail drive shafts as in step 4. of 3360 to 3640 overspeed.
6. Inspect airframe for cracks in area of rotor brake support.
7. Inspect magnetic plugs and strainers in gear boxes for foreign material.
8. Inspect tail drive shaft disconnect for abnormal wear of teeth.

MAXIMUM LEVEL FLIGHT AIRSPEEDS — 72-FOOT MAIN ROTOR AIRSPEEDS LIMITED BY NORMAL POWER OR BLADE STALL TWO-ENGINE OPERATION CONFIGURATION: TWO 150 OR 300 GAL. AUXILIARY TANKS, GEAR UP

RPM	ALTITUDE (FEET) / GROSS WEIGHT (LBS)	SEA LEVEL	2000	4000	6000	8000	10,000
		INDICATED AIRSPEED — KNOTS					
2500	23,500	120	115	112*	103*	93*	80*
	26,000	115	113	103*	92*	77*	56*
	28,500	112	104*	92*	76*	48*	—
	31,000	103	91*	74*	44*		-
2600	23,500	118	116	113	109*	101*	88*
	26,000	116	113	109	100*	85*	64*
	28,500	112	108	100*	84*	56*	—
	31,000	106	99*	82*	52*	—	—

*Denotes blade stall

ENGINES: (2) R2800-54
FUEL GRADE: 115/145

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

Figure 5—8. Maximum Airspeed — 72-Foot Main Rotor

Figure 5—9 deleted.

AIRSPEED LIMITATIONS.

The maximum permissible indicated airspeeds, except those combinations of airspeed and rotor speed at which blade stall is encountered and avoided, are as follows:

FORWARD FLIGHT

Helicopters with 72-foot main rotor (14', 21.5 inch chord blades)	80 knots
Helicopters with 72-foot main rotor (8', 23.65 inch chord blades)	130 knots
Forward Flight (with nose doors open)	40 knots (CAS)
SIDEWARD FLIGHT	30 knots
BACKWARD FLIGHT	20 knots

MANEUVERS AND ACCELERATION.

The following limitations are imposed on maneuvers and acceleration:

a. The angle of bank shall not exceed 30 degrees.

b. Hovering turns shall not be executed at a rate greater than 12 degrees per second (360 degrees in 30 seconds).

c. The flight controls shall not be moved abruptly.

d. The maximum permissible acceleration is 1.5g.

Note

Maneuvers in which rotor blade stall is encountered shall be avoided.

CENTER OF GRAVITY LIMITATIONS.

CG RANGE:

Most Forward cg, sta 227.0 for 68-foot rotor equipped helicopters and sta 228.0 for 72-foot rotor equipped helicopters.

Most Aft cg, sta 245.1.

Station 0 (datum line) is at the nose of the helicopter.

WEIGHT LIMITATIONS.

The maximum recommended gross weight for normal operations is 31000 pounds for 72-foot rotor-equipped helicopters.

CAUTION

It is possible to exceed cg limits if the helicopter is not properly loaded. Check AN 01-1B-

Pages 144 and 145, figure No. 5—10 (sheets 1 and 2) deleted

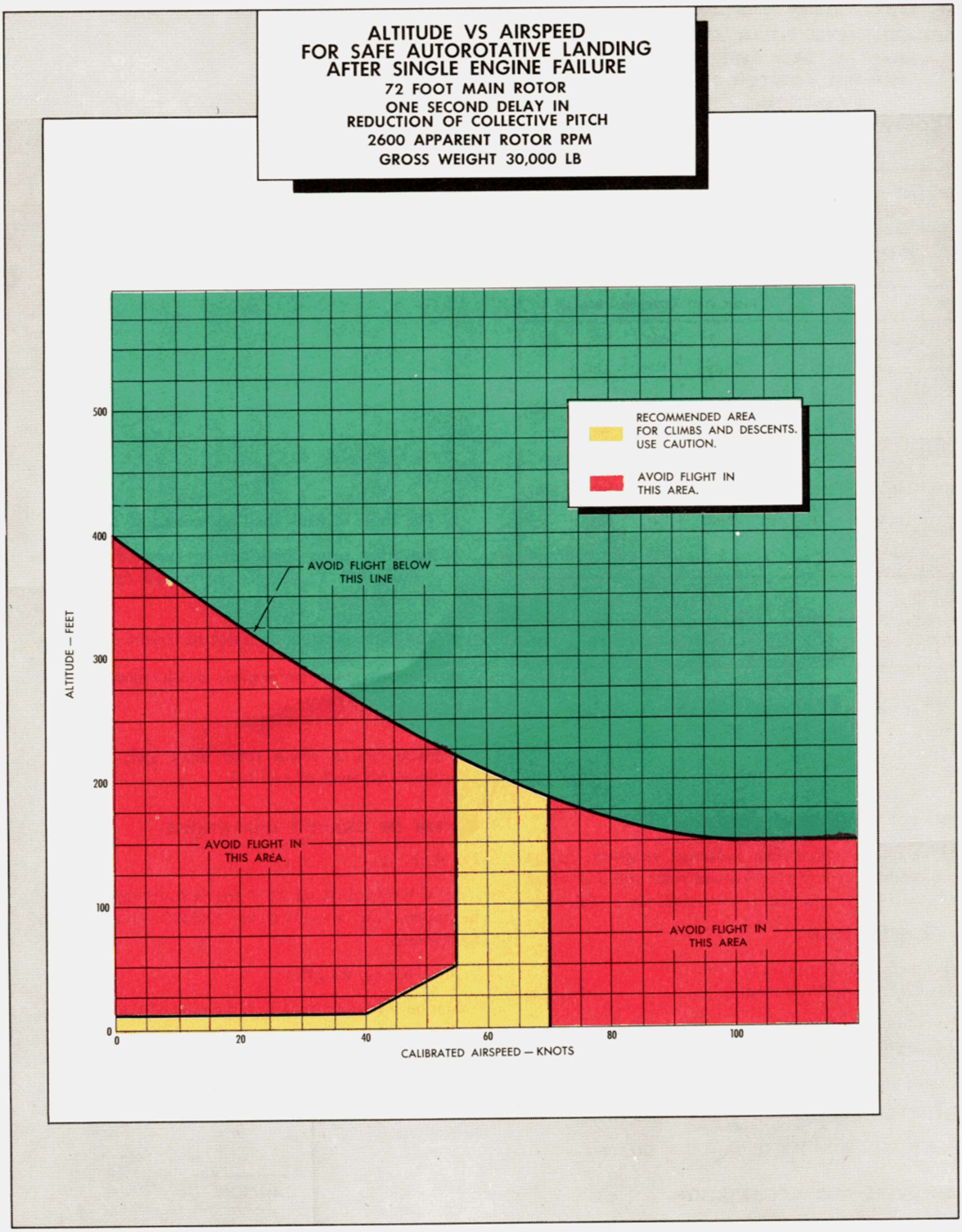


Figure No. 5—11. Area to Avoid for Safe Landing After Power Failure — 72-Foot Main Rotor (Sheet 1)

Figure No. 5–11. Area to Avoid for Safe Landing After Power Failure — 72-Foot Main Rotor (Sheet 2)

40, Handbook of Weight and Balance, and/or Load Adjustor No. R1680-593-3691-GA20, in determining placement of the load for anticipated mission. Use the actual weight of the crew, if available.

CAUTION

Personnel are prohibited from proceeding beyond the aft cabin bulkhead during flight in order not to exceed the center of gravity limitations.

HOVERING LIMITATIONS.

Hovering ceilings out of ground effect may be determined from figure A—20 in Appendix I. The helicopter should not be hovered crosswind when wind velocity exceeds 30 knots.

SECTION VI
FLIGHT CHARACTERISTICS

GENERAL.

The flight characteristics, dependent upon the forward speed and the control surface movement of the fixed wing aircraft, are transferred in a helicopter to the rotational speed and pitch variations of the rotor blades. Since the flight characteristics are independent of forward speed, a helicopter is able to move in any direction at a controlled low speed and still remain safely airborne. The engine-rotor dual tachometers and the manifold pressure gages become the principal flight instruments, especially during hovering and low speed flight, when adequate rotor speed must be kept in order to maintain lift and controllability. The helicopter is directionally stable in forward flight, but in sideward and rearward flight, directional control is more difficult to maintain as the nose of the helicopter has a tendency to turn in the direction of flight. This is due to the location of the center of the fin area which is aft of the main rotor.

POWER SETTLING.

When at high altitudes, at high gross weights, or when operating at reduced power settings, it may not be possible to maintain level flight. Under these circumstances, forward speed is usually low and the helicopter may descend into its own downwash. At certain rates of descent and low forward speeds, this creates an aerodynamic condition called the "vortex ring" state.

The downwash from the rotor begins to recirculate, up, around, and back down through the effective outer rim of the rotor disc. Vortices are built up by the recirculation of air both at the tips of the blades and in the center of the rotor which tends to decrease the effectiveness of the blades.

The resultant turbulent air flow over the blades then results in fuselage and control vibration and sluggish response to control motion in pitching and rolling. In its milder form, it is sometimes confused with carburetor icing or malfunction of the engine. Further application of power (collective pitch) increases the angle of attack of the blades to an excessive degree and increases the rate descent. This is called "power settling."

Recovery is effected by decreasing collective pitch, increasing forward speed, and then increasing power slowly to stop the loss of altitude. An altitude loss of 400 to 700 feet may occur before the condition is recognized and recovery is completed. It is imperative that an entry into the power settling condition be avoided at low altitudes. Some circumstances under which power settling may be expected are:

1. Attempting to hover above the hover ceiling.
2. During a near-vertical approach to a landing when power is added to recover.
3. Flaring in a glide at slow speed and adding power at the wrong instant.
4. Attempting to hover without good ground reference.
5. Trying to recover from a power-on or autorotative descent at zero airspeed by applying power.

BLADE STALL.

During flight conditions, with high values of forward speed, gross weight, power, and altitude, and with low rotor rpm, blade stall may occur at the outer portion of the retreating blade on the left side of the helicopter. Blade stall is indicated by roughness in both the helicopter and flight controls.

The velocity of the air passing over the retreating blade is partially cancelled by the forward speed of the helicopter. The lowering of the relative wind velocity calls

Figure 6—1 deleted.

GENERAL FLIGHT ENVELOPE AT VARIOUS GROSS WEIGHTS

2600 RPM NORMAL MIXTURE 72 FOOT MAIN ROTOR

CONFIGURATION: CLEAN LANDING GEAR UP

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

ENGINES:(2) R-2800-54
FUEL GRADE: 115/145

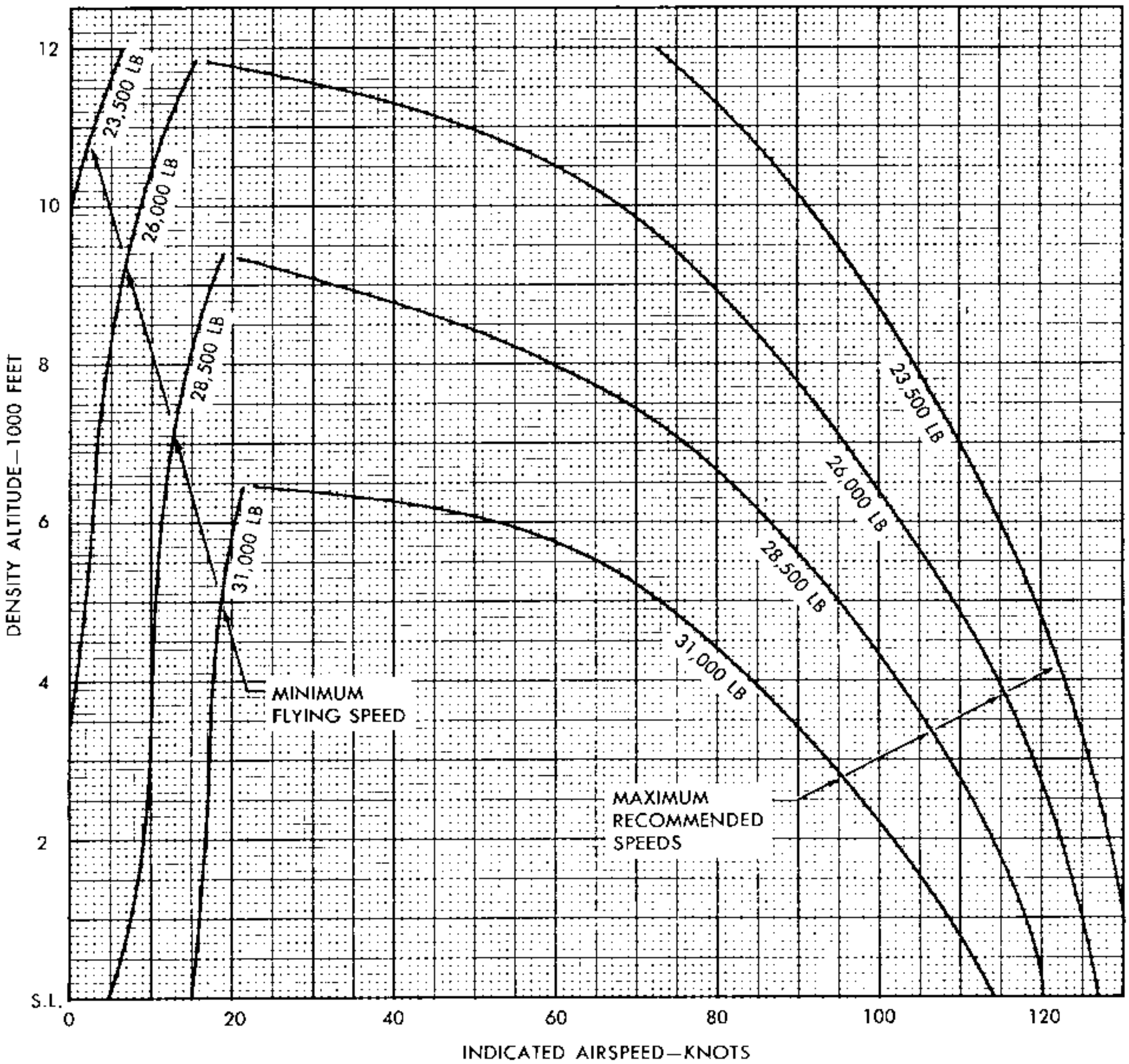


Figure 6—2. General Flight Envelope — 72-Foot Main Rotor

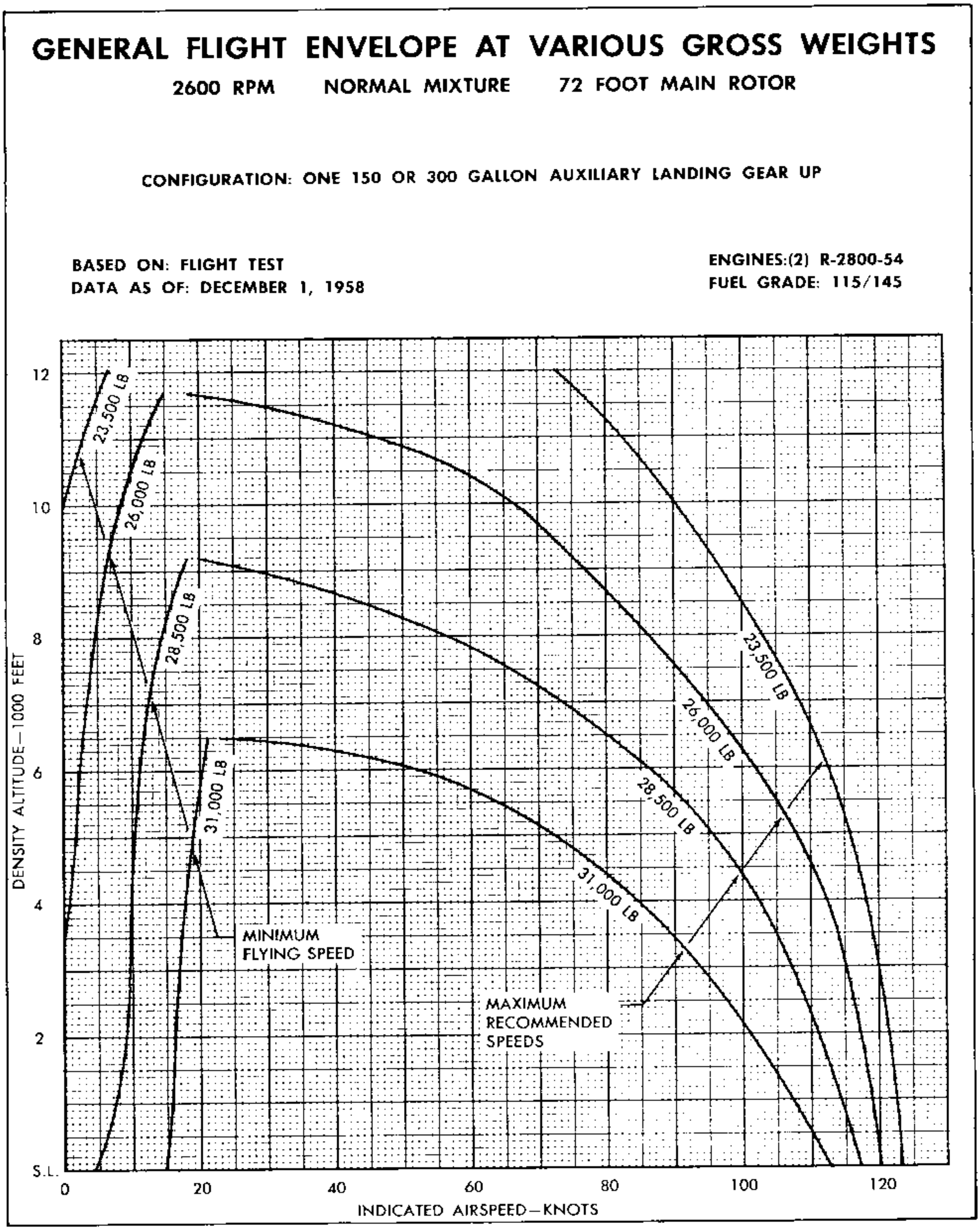


Figure 6–3. General Flight Envelope - 72-Foot Main Rotor

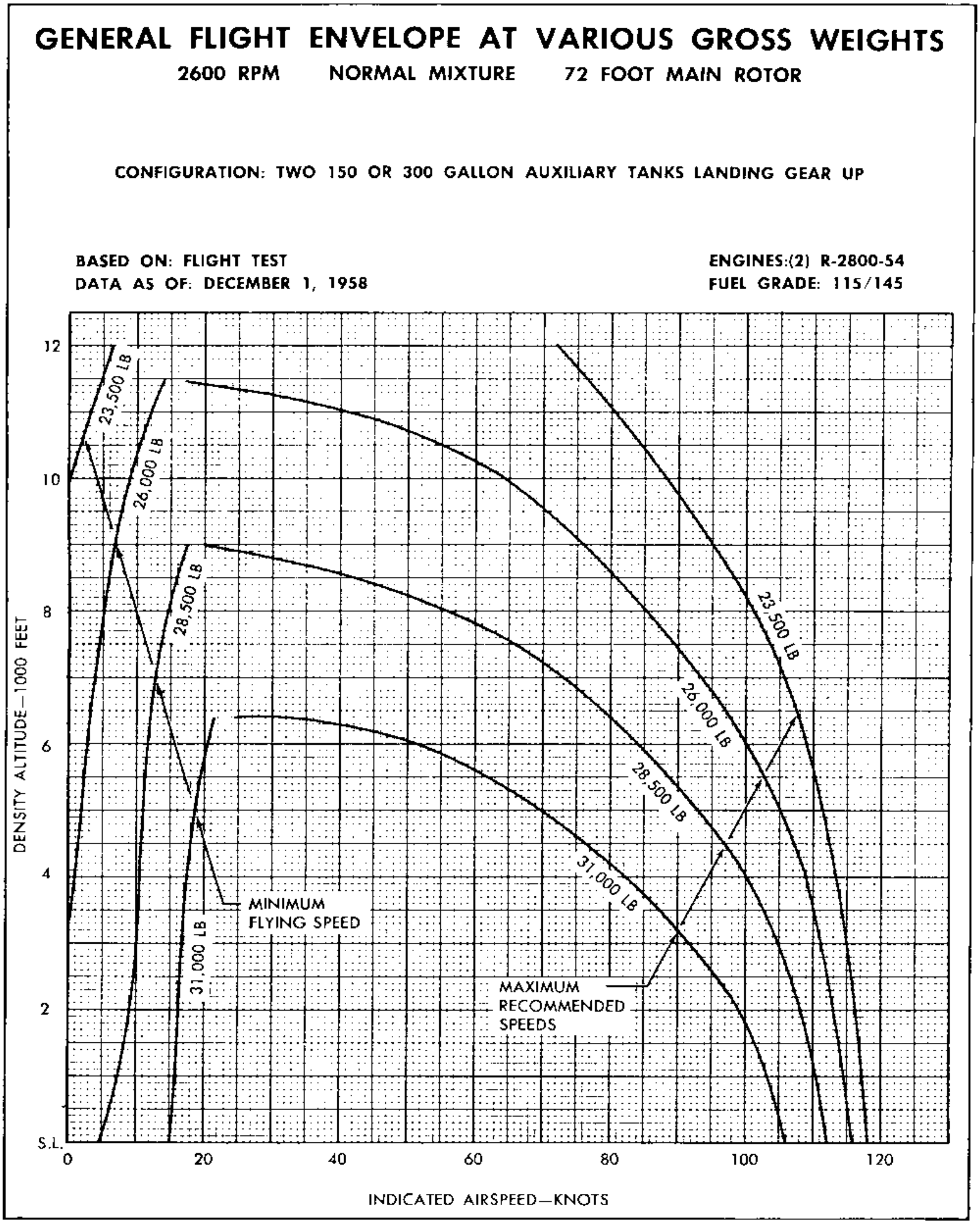


Figure 6-4. General Flight Envelope — 72-Foot Main Rotor

BLADE STALL FOR VARIOUS GROSS WEIGHTS, RPM AND LOAD FACTORS

72 FOOT MAIN ROTOR

CONFIGURATION: CLEAN LANDING GEAR UP

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

ENGINES:(2) R-2800-54
FUEL GRADE: 115/145

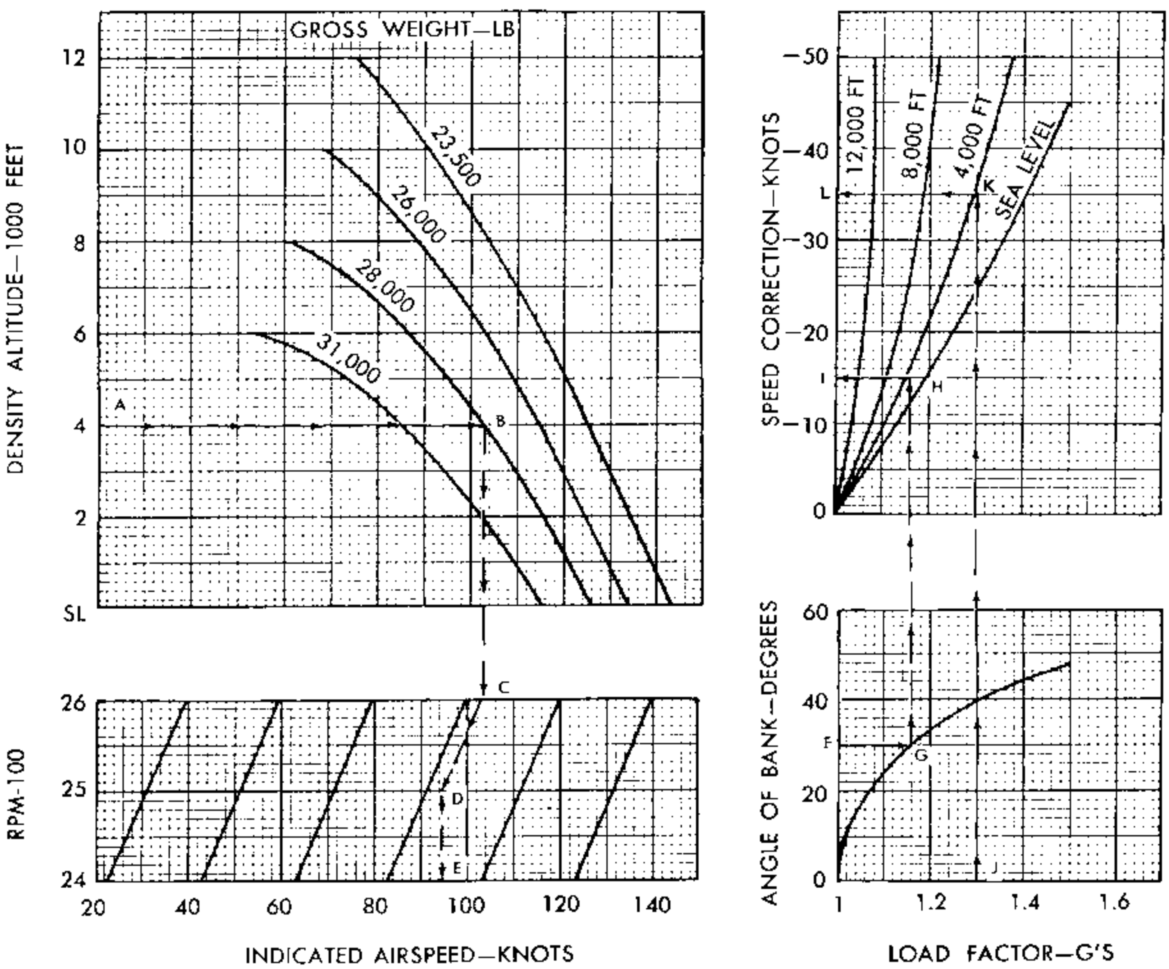


Figure 6–5. Blade Stall Chart – 72-Foot Main Rotor

BLADE STALL FOR VARIOUS GROSS WEIGHTS, RPM AND LOAD FACTORS

72 FOOT MAIN ROTOR

CONFIGURATION: ONE 150 OR 300 GALLON AUXILIARY TANK LANDING GEAR UP

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

ENGINES:(2) R-2800-54
FUEL GRADE: 115/145

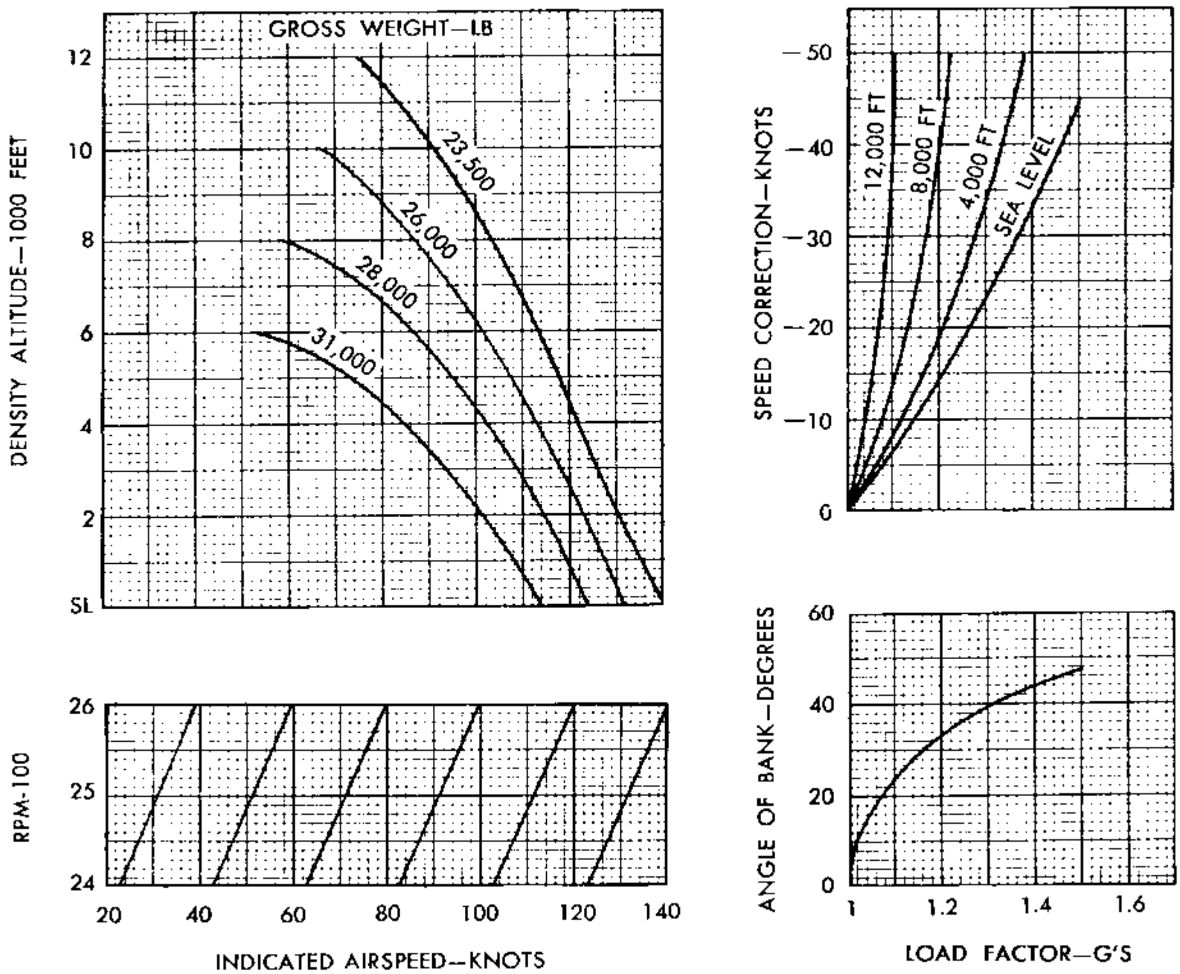


Figure 6—6. Blade Stall Chart — 72-Foot Main Rotor

BLADE STALL FOR VARIOUS GROSS WEIGHTS, RPM AND LOAD FACTORS

72 FOOT MAIN ROTOR

CONFIGURATION: TWO 150 OR 300 GALLON AUXILIARY TANKS LANDING GEAR UP

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

ENGINES:(2) R-2800-54
FUEL GRADE: 115/145

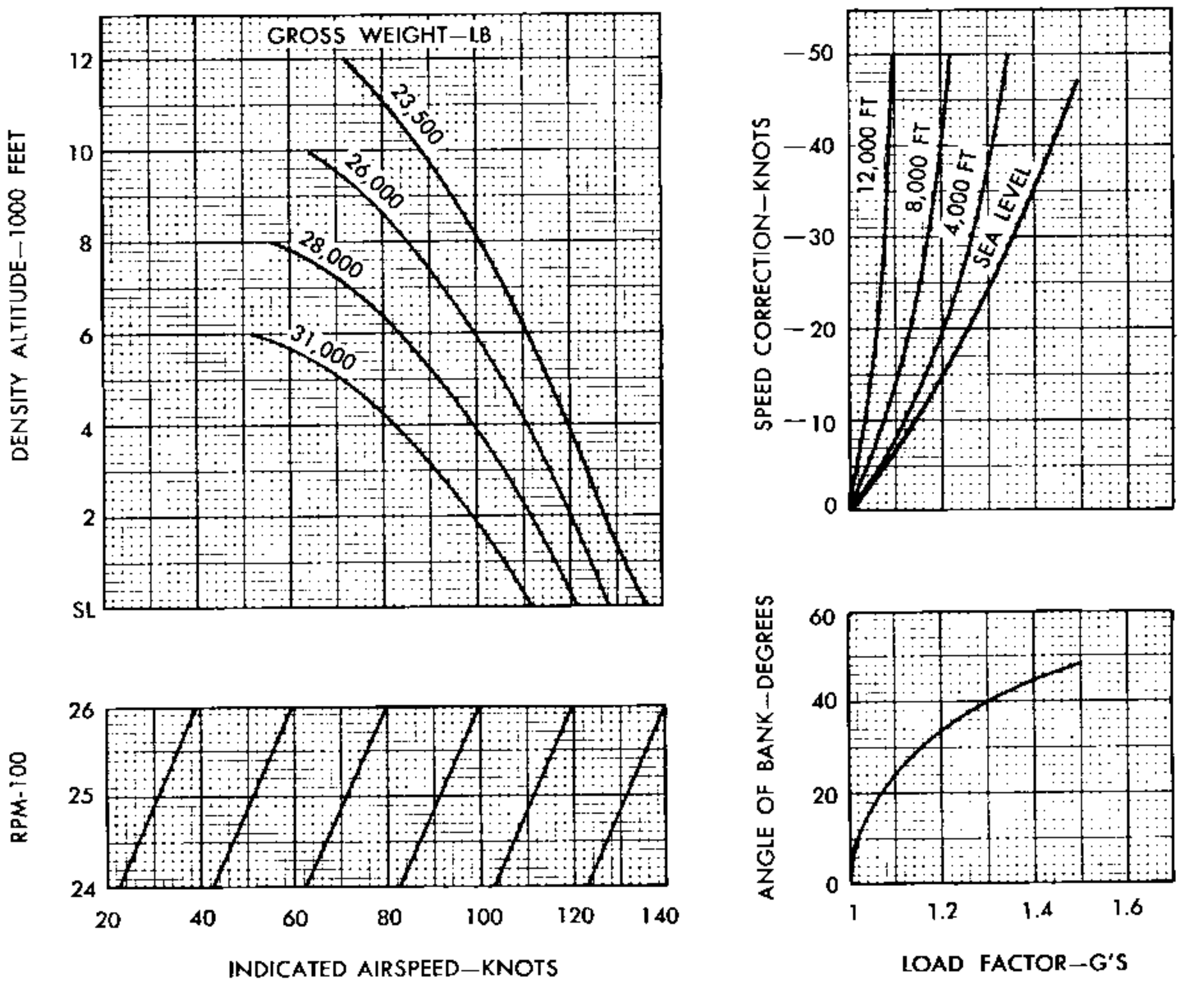


Figure 6–7. Blade Stall Chart — 72-Foot Main Rotor

for a higher angle of attack to maintain the necessary lift. When the angle of attack becomes excessive, blade stall occurs. This blade tip stall is a local condition and exists throughout only a small portion of the rotor disc. It does not easily precipitate a stall over the inner portion of the blade, nor does it spread to other segments of the rotor disc.

Blade stall in straight and level flight does not present any undue hazards as the resulting roughness, felt in the helicopter and controls, provides ample warning. When this roughness occurs, further increases in forward speed, forward application of cyclic control stick, or a reduction in rpm can aggravate blade stall to the point where a reduction in performance and controllability will be experienced.

The stall may be eliminated by accomplishing any one, or a combination of the following:

1. Decrease the airspeed.
2. Increase rotor rpm.
3. Reduce the main rotor pitch.
4. Decrease the severity of the maneuvers.

USE OF THE BLADE STALL CHART.

Example I

Determine the maximum recommended airspeed as limited by blade stall in level flight with a gross weight of 28500 pounds at a density altitude of 4000 feet, using 2500 rpm.

1. Enter the chart (figure 6—5) at a density altitude of 4000 feet (Point A).
2. From Point A, move horizontally to the 28500 pound gross weight curve (Point B).
3. From Point B, proceed vertically to the 2600 rpm baseline (Point C).
4. From Point C, move parallel to the rpm influence lines to 2500 rpm (Point D).
5. From Point D, move vertically to the stalling speed at Point E. The maximum recommended airspeed as limited by blade stall with the above conditions is 95 knots.

Example II

Determine the maximum recommended airspeed due to blade stall in a coordinated turn requiring an angle of bank of 30 degrees with the same gross weight, altitude, and rpm conditions as in Example I.

1. Determine level flight stalling speed as in Example I.
2. Enter the chart (figure 6—5) at an angle of bank of 30 degrees (Point F).
3. From Point F, move horizontally to the load factor curve (Point G).
4. From Point G, move vertically to the airspeed correction line at 4000 feet (Point H).
5. From Point H, move horizontally to the airspeed correction (Point I). For a 30-degree bank angle, the airspeed correction is —15 knots.
6. To determine the maximum recommended airspeed as limited by blade stall during the 30-degree bank, add the airspeed correction of —15 knots to the maximum recommended airspeed in level flight of 95 knots. The corrected airspeed in a coordinated turn requiring a 30-degree bank with the above conditions is 95 —15 = 80 knots.

Example III

Determine the maximum recommended airspeed due to blade stall during a maneuver in which 1.3G's are pulled with the same gross weight, altitude, and rpm conditions as in Example I.

1. Determine level flight stalling speed as in Example I.
2. Enter the chart (figure 6—5) at a load factor of 1.3G's (Point J).
3. From Point J, move vertically to the airspeed correction curve at 4000 feet (Point K).
4. From Point K, move horizontally to the airspeed correction (Point L). With a load factor of 1.3G's, the airspeed correction is —35 knots.
5. To obtain the maximum recommended airspeed as limited by blade stall during a 1.3G maneuver, add the airspeed correction of —35 knots to the maximum recommended airspeed in level flight of 95 knots. The corrected airspeed during 1.3G maneuver with the above conditions is 95 —35 = 60 knots.

FLIGHT CONTROLS.

TWIST-GRIP THROTTLE AND COLLECTIVE PITCH SYNCHRONIZATION. The twist-grip throttle is synchronized with movement of the collective pitch control to retain an approximate constant rpm as collective pitch is increased or decreased. The synchronizer will operate correctly when the throttle grip is prevented from rotating as the collective pitch control is raised or lowered. The pilot may hold the grip or apply a sufficient amount of friction by means of the friction control knob, to prevent the grip from rotating. If the throttle grip is permitted to rotate when changing pitch, the synchronizer will supply insufficient throttle setting changes for the collective pitch control position. Certain loading or atmospheric conditions will also result in improper synchronization. The throttle may then be used independently by rotating the grip. Although the synchronizer will increase power when collective pitch is increased, it is advisable to be prepared to further increase engine power by rotating the twist-grip to the left while increasing pitch to prevent a possible loss of rpm.

During take-off, certain conditions may require full throttle before take-off power is obtained. Full open throttle may be obtained with a small increase in collective pitch at take-off rpm. If collective pitch is increased beyond the point where the throttle is full open, an override spring in the throttle linkage will extend and prevent damage to the system.

FLIGHT CONTROL SERVOS.

The servo units virtually eliminate control forces and the slight forces remaining are constant throughout the full range of control movement. This may cause a tendency to over-control since there is very little "feel" in the cyclic stick unless the stick trim system is in use. If either servo unit should fail or malfunction, it may be turned off provided that there is hydraulic pressure in the other system. Very little difference will be detected by the pilot in the operation of the controls when flying with one servo system inoperative. An automatic stabilization servo system is also installed. Its primary function is to serve as a means of introducing automatic stabilization equipment inputs to the flight controls. The tail rotor control pedal damper and twist-grip throttle servos are also served by the automatic stabilization servo system and all three components will be inoperative when the automatic stabilization servo shut-off switch is placed in the off position.

SERVO CHATTER.

The primary (tandem) flight control servos may develop amplitude limited oscillations (chatter) when they are coupled to automatic blade folding and the strengthened control system. This instability is not self-generating but must be induced by rapid cycling of the control stick. Safety of flight is not involved and the accompanying loads are not in excess of the design limits of the control system, hydraulic servos, and allied components.

CAUTION

External power in excess of 1500 psi must not be applied to either servo system.

There is no reason to check the flight controls prior to flight by violent cycling of the controls. The recommended cycling rate is not to exceed full travel in 1 second. The tandem servo hydraulic systems should be bled of air by full-stroke slow cycling, prior to a flight control check. This is especially true if the helicopter has remained idle for an appreciable length of time. Servos subject to sustained chatter should be checked by maintenance personnel. During engine shutdown procedure, the first stage hydraulic system should be shut off as the rotor head is slowing down, before the blades begin to droop. During engine starting, the first stage hydraulic system should remain shut off until the rotor clutches are engaged.

COORDINATION OF FLIGHT CONTROLS.

The climb and descent of this helicopter is controlled primarily by raising or lowering the collective pitch control; however, coordinated movements of the tail rotor pedals and cyclic control stick are necessary to maintain a constant heading. When collective pitch is increased to ascend, additional torque must be compensated for by depressing the left tail rotor pedal. Sideward flight from hovering is accomplished primarily by lateral displacement of the cyclic control stick; however, it is necessary to use tail rotor control to prevent the nose from swinging toward the direction of flight. In hovering, with no wind, no appreciable movement of the cyclic control stick is necessary; however, with a wind condition, the cyclic control stick should be held into the wind to maintain the same relative position above the ground. Turns, while hovering, are accomplished primarily by depressing the tail rotor pedal corresponding to the desired direction of the turn. During forward flight at low speeds, coordinated movements of the cyclic control stick and tail rotor pedals are necessary to accomplish turns. In high speed flight, less tail rotor pedal displacement is necessary; in fact, turns can be accomplished by lateral movements of the cyclic control stick only.

LEVEL FLIGHT CHARACTERISTICS UNDER VARIOUS SPEED CONDITIONS.

For hovering or low speed flight, high rotor rpm is required because of the high power and control necessary. When, from hovering or low speed flight, increased forward speed is desired, the cyclic control stick is moved forward. A momentary settling may occur with rapid acceleration and then the helicopter will begin to climb because the main rotor blades encounter an increased flow of air due to the forward movement of the helicopter. As the helicopter accelerates to approximately 60 knots, collective pitch and manifold pressure should be steadily decreased to maintain a constant altitude. To maintain the same altitude above approximately 80 knots, an increase in collective pitch and manifold pressure is necessary until maximum speed is reached. At maximum speed, a higher collective pitch setting is required than for hovering. It can be seen that power requirements, therefore, limit the maximum forward speed unless blade stall or compressibility is encountered beforehand. As forward speed is increased, the helicopter will assume an increasing nose down attitude. This is caused by the fact that the main rotor blade flapping hinges are located at a distance from the center of the main rotor hub. When the rotor blade tip-path plane is tilted forward to increase forward speed, the centrifugal force of the blades will tend to align the plane of the rotor hub, and consequently, the fuselage, with the forward tilted tip-path plane. Thus, for high speed flight, a slightly aft cg location will produce a more level fuselage attitude. An aft cg location imposes no limitation on forward speed through lack of forward cyclic control, though normal cyclic control stick operation will be centered at a point further forward.

MANEUVERING FLIGHT.

No acrobatic maneuvers are permitted with this helicopter and normal maneuvers are restricted as shown under **MANEUVERS AND ACCELERATION,** Section V. The freedom of movement, at low speed and normal fuselage attitude, including the ability to take off and land vertically, and to hover over one spot, are the chief advantages of the helicopter.

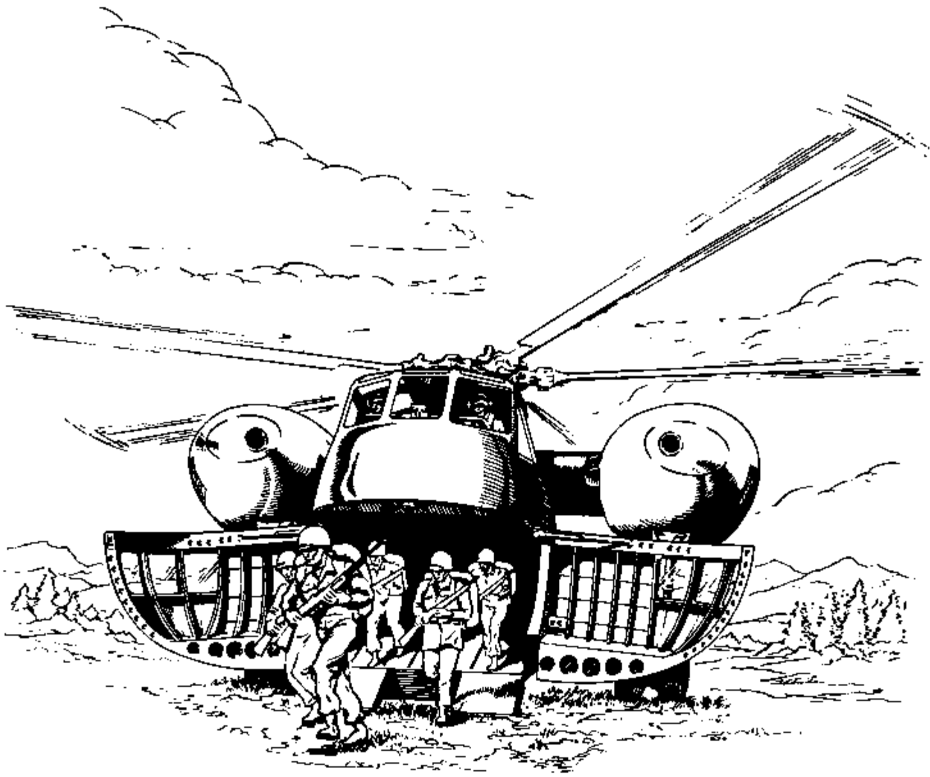
GROUND RESONANCE.

Helicopters equipped with hinged blades and a flexible landing gear are susceptible to some degree of instability during ground operation at certain rotor speeds. This condition, known as ground resonance, may occur at rotor speeds that produce a vibration having the same frequency as the natural frequency of the landing gear. Ground resonance may be aggravated by hard one-wheel landing, by blades that are out of track or are unevenly displaced from their 72-degree relationship by improper operation of the blade dampers, or by improper operation of the landing gear oleo struts. With proper technique and maintenance, ground resonance should not occur with this helicopter because of the design of the rotor head and landing gear. However, if ground resonance should occur, corrective action must be taken immediately or severe oscillations will build up so rapidly that damage to the helicopter will result.

CAUTION

If ground resonance should occur with high rpm, immediately increase power and take off. The resonance will stop when the helicopter becomes airborne. If ground resonance should occur at a low rpm, immediately decrease rotor rpm further by decreasing throttle and apply both the rotor brake lever and the wheel brakes.

To prevent the possibility of ground resonance occurring during take-off, steadily increase collective pitch until the wheels are clear of the ground; during landings, steadily decrease collective pitch as the wheels contact the ground.



SECTION VII
SYSTEMS OPERATION

ENGINES.

MIXTURE LEVER.

Manual leaning beyond the NORMAL detent position is not recommended while the rotor is engaged. The RICH position of the carburetor mixture control levers should be used for take-off, landing, and when high power is required. The NORMAL position of the mixture control levers should be used at all other times as long as the cylinder head temperature remains within limits and engine operation is normal.

CARBURETOR ICE.

Carburetor ice is of three different types: impact ice, formed by snow, sleet, or sub-cooled water in the atmosphere; throttle ice, formed on or near the throttle because of the cooling effect caused by restrictions in the carburetor throat or air passages; and ice formed by the cooling effect due to evaporation of fuel as it is sprayed into the blower throat. Indications of carburetor icing are: a decrease in manifold pressure resulting in loss of power, changes in fuel-air ratio, either richer or leaner as indicated by rough engine operation; uneven response of manifold pressure to throttle setting changes due to ice jamming or sticking the carburetor throttle, and icing on other components of the helicopter. To prevent the formation of carburetor ice, set the mixture control in RICH and maintain sufficient carburetor heat to keep the carburetor air temperature within the normal operating range. Removal of ice already formed is best accomplished by use of full carburetor heat. Other methods of eliminating ice are to set the mixture RICH, increase the throttle, or change altitude to ice-free atmospheric conditions.

BACKFIRING.

After a cold engine is started, the throttle should be increased slightly to produce an initial idling speed of 1000 rpm. If the throttle is opened too wide, it should be closed quickly as the excessive amount of air admitted to the cold engine rapidly reduces the fuel-air mixture and may cause backfiring. Backfiring may also be caused by underpriming.

Note

> Do not pump the throttle after starting while the engine is still cold as it frequently causes backfiring and may cause serious damage to the induction system. It also presents a fire hazard.

SPARK PLUG FOULING.

Spark plug fouling is a principle cause of ignition trouble, which in turn is one of the most common engine maintenance and operating problems with aircraft engines using 115/145 or 100/130 grade fuel. These grades of fuel may contain a relatively high lead content, up to 4.6 cc. Such fouling might be defined as an accumulation of deposits which cause misfiring or prevent firing across the spark plug electrodes. The most common types of fouling are lead fouling and carbon fouling, with lead fouling the main troublemaker. Cause, prevention, and cure of spark plug fouling are all linked to the chemistry and physics of the combustion cycle, which in turn are subject to wide variation under different ground and flight engine operating conditions. A logical treatment of the problem involves separate discussion of each aspect of typical engine operation, including ground running, take-off, climb, cruise, and descent, following the typical sequence of operation in the Flight manual. Prevention appears to be the most profitable line of attack on the problem.

Tetraethyl lead is the most important basic cause of lead fouling. Scavenger agents such as bromine in the tetraethyl lead are provided to combine with the lead during combustion, removing it with the exhaust gases. However, under certain conditions of temperature and pressure, the lead will condense out on the spark plug insulator as lead oxide or lead bromide compounds. In the presence of excess carbon as a reducing agent, these may form metallic lead particles. All such deposits can prevent ignition or firing. Obviously, the best solution is to remove or reduce the lead compounds presently contaminating the fuels. Other pertinent factors which influence plug misfiring include the type of ignition system, spark plug characteristics, and general engine conditioning, including the care and handling of spark plugs, the operating requirements and characteristics of the particular engine installation, and the specific engine operating conditions.

In general, spark plug fouling involves a build-up of deposits through prolonged operation under a fixed set of conditions. Prevention and remedy for plug fouling, therefore, depend on taking action to vary these conditions, upset the chemistry of the fouling cycle, and restore good ignition.

GROUND RUNNING.

1. Type and Cause: Either lead or carbon fouling. Lead fouling may be residual from a previous flight. Carbon fouling is usually due to prolonged ground running at idle, particularly when the idle mixture is richer than best power; excess carbon from the rich mixture plus engine oil in combustion tend to build up as fouling deposits. The symptoms of such fouling include excessive rpm drop at the ignition system check at field barometric manifold pressure.
2. Prevention: The importance of preventing plug fouling during ground running is obvious, since this type of fouling is apt to cause flight delays either as a result of excessive rpm drops during the ignition system check or due to engine malfunction during take-off in frequent cases where the plug fouling has occurred, but is not apparent at any stage of the engine check-out procedure. Wherever possible, avoid prolonged or unnecessary ground running. The idle mixture should be adjusted to best power mixture at the idle speed commonly used for ground running, rather than at the minimum idle speed, since there is a tendency for the mixture to enrich with any increase in rpm, and excessively rich idling mixture is the most common cause of carbon fouling. It is important to properly adjust idle mixture, and to frequently check to insure the setting is correct. After each 10 minutes of ground running with the rotor engaged, the engine should be run up to field barometic manifold pressure for 1 minute.
3. Cure: A cure for fouling is less dependable than adequate preventive procedures, and the only practical cure may be a spark plug change: Generally, the preventive action is the best cure. The 1 minute run at barometric manifold pressure with the rotor engaged may, however, have a curative effect. Additional running for short periods of time, with manifold pressure changes of about 2 inches Hg, is sometimes effective. Due to generally poor engine cooling under such conditions, caution must be used to avoid any prolonged operation at or above field barometric manifold pressure.

TAKE-OFF.

1. Type and Cause: Lead and/or carbon fouling. The rapid change in combustion temperatures and pressures and the high levels achieved under take-off conditions, are favorable to spark plug misfiring if there is any fouling from previous flight or ground running. The electrical resistance of residual deposits decreases rapidly as limiting temperatures are approached, so that the spark may short circuit along the insulator rather than firing the gap. If excess carbon is present, metallic lead may be formed by reducing action of free carbon on lead oxides and lead bromides. The most common symptoms include power loss and erratic power fluctuations and associated backfiring and rough running; all of these are evidences of advanced stages of misfiring. Misfiring of a few plugs will normally go undetected.
2. Prevention: Same as for ground running. In addition, it is important to reduce cylinder head temperature to the recommended pretake-off cylinder head temperature level to take advantage of the increased BHP, and decreased tendency for misfiring with relatively cool cylinder head temperature during take-off. Smooth and steady application of throttle is preferable to rapid or "jam" acceleration to take-off power.
3. Cure: Reduce manifold pressure 2 to 5 inches Hg or as required to restore smooth operation.

CLIMB.

Fouling is rarely a problem during climb. It might persist as a continuation of take-off fouling. A reduction in manifold pressure will tend to restore normal ignition.

CRUISE.

1. Type and Cause: Cruise conditions usually generate lead fouling rather than carbon fouling. Conditions favorable to lead fouling include long-continued application of a given set of engine conditions typical of cruise flight, particularly those involving lean mixtures, at or leaner than NORMAL posi-

tion. Associated contributing factors include abnormally cool cylinder head temperature and low brake mean effective pressure. High tension ignition appears more susceptible to fouling than low tension. Common symptoms include backfiring or afterfiring.

2. Prevention: Although a periodic change in engine conditions will usually forestall lead fouling, the nature of helicopter cruise operation minimizes this type of fouling. Hourly intervals are frequently used, and may accomplish results either by use of rich mixture for 5 minutes; use of primer for 1 minute; a change of manifold pressure of 3 to 5 inches Hg, or a change of 100 to 130 rpm. A reduction in power level followed by an increase appears to be the preferable approach to prevention. One or more of these procedures should prove effective, and the procedure which least affects the flight condition is naturally preferable.
3. Cure: Cure is less certain and includes a wider variety of procedures than prevention. One technique involves a complete change in the power cycle, including use of rich mixture, a reduction of manifold pressure of 8 to 10 inches Hg, and a period of engine cooling under low output conditions, followed by gradual restoration of cruise power in increments of 2 to 3 inches Hg with several minutes of operation at each level. Generally, plugs which are misfiring or completely fouled are apt to resume firing at lower power settings; therefore, it is preferable to reduce power and then restore it, rather than attempt to reach a high power level with malfunctioning plugs. High power burn-out procedures may change the chemistry of the fouling deposits to the extent that a change of spark plugs is the only cure. They also introduce the possibility of destructive backfiring during application of increased power.

DESCENT.

1. Type and Cause: Plug fouling in descent may be carbon or lead fouling, favored by low power output conditions such as cool cylinder head temperature low BMEP and cold carburetor air temperature often associated with descent.

This type of plug fouling is minimized by the relatively short duration of helicopter descents.

CYLINDER HEAD TEMPERATURE.

To keep cylinder head materials at high operating strength, it is recommended that the cylinder head temperature be held within the green arc as shown in figure 5—1. Cylinder head temperature may be controlled by regulating the carburetor air lever. Other means of controlling cylinder head temperature are by use of the mixture control lever and throttle. A rich mixture will produce cooler operating temperatures than a lean mixture; a decrease in manifold pressure will also produce lower operating temperatures.

DETONATION.

Normal combustion produces a rapid but even burning of the fuel-air mixture in the combustion chamber. Detonation, on the other hand, is a sudden violent explosion which transmits extremely high pressures and impact loads to the pistons, cylinders, and other components of the engine. Detonation produces a loss in power and causes a rapid increase in cylinder head temperature which may result in damage or failure of the engine. The increase in cylinder head temperature may cause a hot spot in the combustion chamber, which will ignite the fuel-air charge far in advance of the spark timing. This condition is known as pre-ignition and will further increase engine temperature. Conditions causing detonation are excessive manifold pressure, excessive cylinder head temperature, excessive carburetor air temperature, advance spark timing, lean mixtures, or grades of fuel with lower anti-knock rating than is required by the engine. Detonation may be eliminated by decreasing manifold pressure and rpm or by enriching mixture and decreasing carburetor air temperature.

HYDRAULIC LOCK.

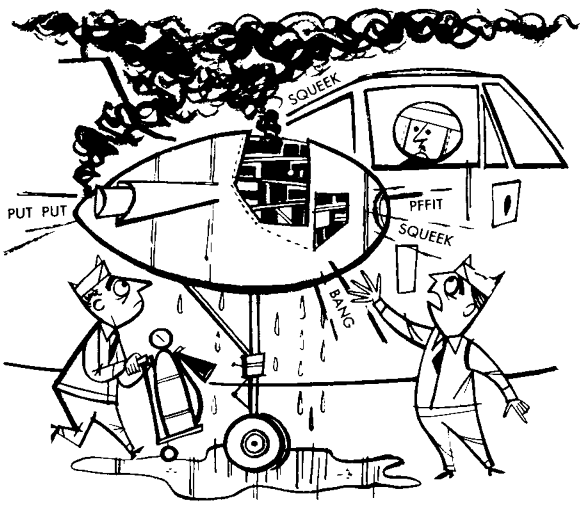
If the engine has not been run for 2 hours or more, or if it is suspected that excessive fuel and oil have drained into the lower cylinders, turn the engine through several revolutions by engaging the starter for a period of 30 seconds to clear the lower cylinders. If the starter should suddenly slow down or hesitate when turning the engine over, or if the engine should stop and the starter speed up because of a slipping starter clutch, the pistons are being forced against unusually high pressures. This is an indication of hydraulic lock. If turning over of the engine is continued, damage to the engine may result. When hydraulic lock is indicated, the spark plugs should be removed and the fluid allowed to drain from the cylinders before attempting to start. A ground crew member should stand by each engine throughout the starting operation to assure that the engine turns over on the starter, to check for leaks or erratic operation, and to act as fire guard.

PRIMING.

Starting should be accomplished by priming. The fuel booster pump must be used to supply the fuel pressure necessary for priming. The priming fuel is sprayed into the induction system below the carburetor where it is mixed with air from the carburetor and carried into the intake ports of the cylinders. The amount of priming necessary for starting depends on air and engine temperatures. Under exceedingly cold atmospheric and engine conditions, it may be necessary to actuate the primer continuously to affect an engine start. Conversely, under temperature or tropical conditions, intermittent prime should be used and caution exercised to avoid over-priming.

OVERPRIMING.

Overpriming will load the cylinders with fuel and make the engine difficult to start. An indication of over-priming is weak firing, followed by dense black smoke. In extreme cases of overpriming, the fuel will wash the oil film from the cylinder walls, pistons, and piston rings. Without this lubricating oil film, there is danger of scoring the cylinder walls and piston seizure. This condition will be indicated by squeaking of the pistons when the engine is being turned over by the starter. Do not attempt to start the engine, when this squeaking is heard, until the spark plugs have been removed and the cylinder walls sprayed with fresh oil. When the engine has been slightly overprimed and will not start, and the squeaking is not heard or hydraulic lock is not indicated, the engine may be turned over by the starter for several revolutions with the ignition switch OFF and the mixture control in IDLE CUT-OFF. Repeat the starting procedure with little or no priming.



"I GUESS IT'S OVERPRIMED!"

UNDERPRIMING.

Indications of underpriming are failure of the engine to start, erratic running, weak firing with insufficient energy to turn the engine over, puffs of white smoke, and backfiring. When any of the above conditions occur, check that the fuel shut-off switch is properly set and that there is sufficient fuel pressure for priming. Additional priming should be done cautiously.

ROTOR CLUTCHES.

Each of the hydro-mechanical clutches which provide the engagement between the engines and the transmission system, as described in Section I, is composed of a fluid coupling and a mechanical coupling incorporating a freewheeling unit. The fluid coupling consists of two sets of vanes enclosed in a housing. The vanes face each other and rotate about the same axis but are not connected to each other. One set of vanes, the driving vanes, is attached to the engine shaft and rotates at engine speed. The second set, the driven vanes, is attached to the main transmission drive shaft. The mechanical coupling consists of a freewheeling unit, a fly-weight assembly, and a blocker plate. To engage the transmission and rotor systems, the rotor clutch pump switches are turned on when the engines are operating at 1400 to 1500 rpm. Oil will be pumped into the fluid coupling housings where it is impelled by the driving vanes. This motion is transmitted to the driven vanes which will begin to drive the rotor system. Engine speed is increased slowly to 2100 rpm with the collective pitch control in the low pitch position. As the transmission system accelerates, as indicated by the rotor pointer of the dual tachometer, an increase in throttle will be necessary to maintain 2100 rpm. During the above procedure and as long as engine rpm remains greater than the equivalent main rotor rpm, the blocker plates in the mechanical couplings prevent direct mechanical engagement. When main rotor speed has increased to approximately 1400 apparent rpm, the quadrant throttle of one engine is decreased to drop engine rpm below the corresponding main rotor rpm. The decrease in engine rpm allows the blocker plate to slide out of position, releasing the flyweights. The centrifugal force of the flyweights places the free-wheeling unit in operation. To complete the direct mechanical drive, engine rpm is increased until the engine pointer and the main rotor pointer of the dual tachometers are in line. As the throttle is increased to complete the mechanical drive of one engine, the throttle of the second engine is backed off to prevent over-speeding as the direct drive of one engine will be carrying the entire transmission driving load. The throttle should be decreased on the second engine a sufficient amount to allow the engine tachometer pointer to drop below the rotor tachometer pointer. After the pointer of the second engine tachometer has fallen below the rotor pointer, the quadrant throttle of the second engine is then advanced to increase rpm and complete its direct mechanical drive. After the mechanical engagement of both clutches is completed, the clutch pumps are turned OFF and the oil in the housing of the fluid couplings will drain back into the clutch oil tanks. As long as rotor speed is held above approximately 1120 apparent rpm, the mechanical couplings will remain engaged although they may be either in the freewheeling position or in the driving position. The clutches are

OPERATION OF FUEL SYSTEM				
	CONDITION	CROSS-FEED SWITCH	FUEL SHUT-OFF VALVES	BOOSTER PUMP SWITCHES
BOTH ENGINES OPERATING	Normal Operotion — Left tank to left engine and right tank to right engine.	CLOSED	Both — ON	Both — NORM
	Both tanks to both engines.	OPEN	Both — ON	Both — NORM
	Either tank to both engines.	OPEN	Both — ON	Tank in use — EMER Tank not in use — OFF
ONE ENGINE OPERATING	Left tonk to left engine or right tonk to right engine.	CLOSED	Good engine — ON Failed engine — OFF	Good engine — NORM Failed engine — OFF
	Both tanks to either engine.	OPEN	Good engine — ON Failed engine — OFF	Both — NORM
	Either tank to opposite engine.	OPEN	Good engine — ON Failed engine — OFF	Tank in use — NORM Tank not in use — OFF

Figure 7—1. Fuel Management

disengaged by closing the throttles and permitting main rotor speed to drop below 1120 apparent rpm. A spring in the mechanical couplings will override the centrifugal force created by the flyweights at this rpm and push the mechanical couplings to the disengaged position.

FUEL SYSTEM MANAGEMENT.

For normal operation, the left fuel tank supplies fuel to the left engine and the right fuel tank supplies the right engine. For operation under unusual conditions, the fuel system controls, together with the fuel cross-feed system, provide such flexibility of operation that the table in figure 7—1 is included as a guide. When utilizing the cross-feed system (both engines from one tank), do not place the fuel shut-off switch for the unused system at OFF. The valve it controls is downstream of the cross-feed valve and will shut off fuel supply to that engine.

During starting, ground operations, and take-off, only the main tanks are used. The auxiliary fuel tank pumps are checked for operation prior to take-off and are placed in operation after take-off. It is advisable to commence transfer as soon as possible to insure full main tanks and decrease the severity of a symmetrical loading should the auxiliary fuel tank pumps fail.

During single engine operation, fuel may be used from both tanks at the same time or from one tank at a time. When using fuel from both tanks, it is possible that fuel will actually be supplied from one tank only. This can occur if the difference in the normal operating pressure of the fuel booster pumps is sufficient to close the check valve downstream of the weaker pump. When using fuel from both tanks, check the fuel quantity gages periodically. If fuel is being consumed at an unequal rate from the tanks, it may be more satisfactory to operate from one tank at a time in order to equalize the fuel quantity in each tank. When changing from two-tank to single-tank operation, check the fuel pressure gage of the engine in operation (the fuel pressure gage of the shut-down engine will be shut off). If the fuel pressure drops to the lower range marking limit, place the booster pump switch for the pump in operation in the EMER position.

FUEL BOOSTER PUMP OPERATION.

Fuel booster pumps are located in the sumps of the right and left wing tanks and supply fuel under pressure to the two independent fuel systems. Normally, fuel flows from the wing tanks through fuel strainers and shut-off valves to engine-driven fuel pumps and then to the carburetors. Should an engine-driven fuel pump fail, the booster pump will deliver sufficient fuel pressure to the carburetor for proper operation. The NORM position of the booster pump switch is used during engine start, warm-up, and during flight. The EMER position is used for take-off, approach, landing, and if an engine-driven fuel pump fails. If the fuel pressure fluctuates while in the NORM position when flying at altitude, the switch should be placed in the EMER position. During cross-feed operations, when one fuel tank is being used to supply fuel to both engines, the EMER position should be used.

QUADRANT THROTTLE AND TWIST-GRIP THROTTLE OPERATION.

The following information is supplemental to the description of the two throttles in Section I and their operation in Section II. The quadrant throttles are connected directly to the carburetors and the position of the quadrant throttle levers is a constant indication of actual carburetor throttle position. The twist-grip throttles on the collective pitch control will operate the throttle at the carburetors only when not restricted by the quadrant throttles. When the twist-grip throttles are moved to control the engines, the quadrant throttle levers will move accordingly; however, when the quadrant throttles are moved, the motion is not transmitted to the twist-grips.

When starting the engines, the twist-grip throttles are preset to 40 degrees as indicated on the twist-grip throttle indicator with the quadrant throttle levers locked closed. The preset twist-grip position does not preset the carburetor throttles because the quadrant throttle levers are locked closed. The quadrant throttles in the closed and locked position will provide enough throttle for engine starting. If the quadrant throttles are opened, microswitches on the front of the quadrant will open and cut off direct current from the engine starting systems. During warm-up, each engine is controlled by its own quadrant throttle. The levers operate smoothly with no detents or ratchets. The clutches are engaged with the quadrant throttles at approximately 10 to 12 degrees which corresponds to the 40-degree setting of the twist-grip throttles, and the manifold pressure of the engines is then equalized at approximately 2500 rpm. Fine synchronizing adjustments can be made by adjusting the levers up or down while watching the manifold pressure gages. When making larger synchronizing adjustments, it is advisable to move one throttle lever up or down in order not to disturb the relation between the twist-grip throttles and the quadrant throttles. The engines may be synchronized before clutch engagement if extensive ground operation with both engines running is desired, in which case they will have to be synchronized again after the clutches are engaged. After clutch engagement and synchronization, the quadrant throttles are left alone and control of both engines is accomplished simultaneously by means of either twist-grip throttle. When collective pitch is increased, the automatic synchronization between throttle action and collective pitch will operate to increase the throttle, corrections being made to maintain or increase rpm for take-off by rotating the twist-grip slightly as the collective pitch control is raised. The quadrant throttles should not be used while the helicopter is being flown with the twist-grip throttles except for fine engine synchronization adjustments while the twist-grip and collective pitch control are held steady. To move both quadrant throttles while in flight, or to equalize engine manifold pressure by always moving the quadrant throttles in one direction only, would throw the synchronization between the twist-grip throttles and the collective pitch control out of adjustment, resulting in the necessity of rotating the twist-grip excessively as the collective pitch control is moved. This could also result in lack of authority of the twist-grip throttle to command full power for landing or minimum power for engine shutdown, as it would be rotated against either of its stops with the quadrant throttles still far from their stops. At normal landing approach powers, the twist-grip throttle indicator should read within the range of 10 to 70 degrees. If the indicator reads above 70 degrees with low engine power, the pilot should back off the twist-grip throttle and the copilot should advance both control quadrant throttles, maintaining constant engine power, until the indicator reads 40 degrees. If the indicator reads below 10 degrees with high engine power, the pilot should advance the twist-grip throttle and the copilot should back off both quadrant throttles until the indicator reads 40 degrees. Maintain steady engine rpm during these adjustments. The final position of the twist-grip indicator should bear approximately the same relationship to its total range of travel as the position of the quadrant throttles do to their total range of travel. Whenever loss of one engine is experienced, do not place its respective quadrant throttle in the CLOSED position. The frictional forces would be increased thereby, and difficulty experienced in moving the twist-grip throttle should the throttle be rendered inoperative.

AUTOMATIC STABILIZATION EQUIPMENT (ASE).

A more detailed account of the system is covered in the following paragraphs:

In addition to the control panel and the push buttons on the cyclic control stick grips, the major components of the system consist of a vertical gyro and an amplifier located port and forward of the main gear box. A barometric altitude control is located overhead on the port side of the main cabin. Four servo motors that control altitude, yaw, roll, and pitch are part of the automatic stabilization servo system, located behind the pilot's seat. A servo adjustment box is located adjacent to the servo assembly.

The sensing signals for the system, furnished by the elements listed below, are fed directly into the amplifier, except the yaw signals which first pass through the control panel. The yaw signals, furnished by the G.E. Type MA-1 gyro-magnetic compass system, are modified by adjustment of the yaw trim knob in the control panel and are then fed into the amplifier. The pitch and roll signals are provided by a vertical gyro. The barometric altitude signals are provided by the barometric altitude control which is connected to the static port.

In the amplifier, rate signals associated with the sensing signals are electronically derived, mixed with the sensing signals, and amplified. These signals are fed to the servo adjustment box and then fed as input signals to drive the servo motors. The servo motors actuate the pilot valves which, in turn, control the power pistons of the automatic stabilization servo system which actuate the flight controls. Attached to the servo motors are small displacement transducers which provide voltages proportional to the servo motor outputs. These voltages are fed back into the amplifier through the servo adjustment box and any errors detected between them and the sensing signals are constantly reintroduced to the servo motors to smooth out the automatic stabilization equipment corrective action and prevent excessive overshooting.

With the system engaged, the stabilized behavior of the helicopter is controlled by conventional use of the flight controls. Pitch and roll stick command pick-offs, located in the control linkage forward of the auxiliary servo, sense the motions of the cyclic stick and feed proportional voltage signals into the amplifier whenever the stick is moved. The vertical gyro senses the pilot-induced attitude change and also sends a signal to the amplifier. The stick command pick-off signals, in effect, null this signal from the vertical gyro and retain the helicopter in the new attitude corresponding to the new stick position.

To command the system in yaw in order to execute turns, the tail rotor pedals may be used in the conventional manner. Directional control cut-out microswitches, mounted on the four tail rotor control pedals, are connected in series. Whenever the feet are on the pedals, this switch circuit is opened and no directional sensing signal is fed into the amplifier. Simultaneously, a circuit in the yaw channel of the amplifier is closed, which nulls the yaw signal. After the desired heading has been assumed, and as soon as the feet are removed from the pedals, the directional sensing signal is again fed into the amplifier and the nulling circuit is opened. The helicopter will then stabilize on the new heading and will not return to the original heading. Heading changes may also be made by turning the yaw trim knob on the control panel.

With the barometric altitude mode in operation, the collective pitch control may be used to override the automatic stabilization equipment; however, when an altitude change is desired, the altitude mode should be disengaged, the helicopter brought to the new altitude, and the altitude control re-engaged. If the BAR ALT button is not disengaged when changing altitude, the helicopter will attempt to remain at the altitude at which the BAR ALT button was depressed.

In addition to the hard-over switch, the servo adjustment box is fitted with a selector switch, which permits use of the null indicator on the control panel to monitor the motion of all servo motors. Normally, it is kept in the PITCH position so that the indicator can be used with the cg trim knob to adjust for shifting cg. A spring-loaded switch, marked MOTOR, located on the servo adjustment box, when actuated to the INPUT position transforms the null indicator into a voltmeter in order to check the input error voltages to the servo motors. Also accessible are the servo adjustment test points and the null adjustment potentiometers. They are set when leaving the factory and, except for system alignment or overhaul, should not be disturbed. Other switches on the servo adjustment box are: a COM CHECK switch, used to perform a maintenance check on the dual windings of the stick command pick-off; a spring-loaded DELAY OVERRIDE switch, used to put the automatic stabilization equipment in standby prior to completing the 3-minute warm-up period for maintenance purposes; and two XDCER CHECK buttons, labeled L and R, used to test for any failure of either of the dual windings of the servo motor transducer.

SECTION VIII
CREW DUTIES

GENERAL.

The basic crew required to fly the helicopter is one pilot and one copilot. Primary and alternate duties of each are discussed, although normal procedures as given in Section II will not be repeated. Duties are also suggested for a crew member in the cabin.

PILOT.

The pilot has over-all responsibility for the helicopter and the efficient completion of the mission. He should arrange to have all persons briefed on normal and emergency procedures and correct use of equipment. Before each flight, especially if troops or litter patients are carried, one man should be appointed to be in charge of the cabin. The helicopter may be flown completely by the pilot, although flight controls and flight instruments are duplicated for the copilot. All circuit breakers, switches, and controls on the overhead panels and the console and the engine control quadrant are accessible to the pilot and the copilot. The radio controls, except the PRC liaison set, which is at the pilot's right, are equally accessible to the pilot or copilot. The arrangement of controls in the pilot's compartment is such that in addition to flying the helicopter, the pilot will operate the following equipment:

1. Gyro-Magnetic Compass Control Panel.
2. Radar Height Indicator.
3. Stabilizer Positioning Switch.
4. Wheel Brakes and Parking Brake Handle.
5. Rotor Brake Lever.
6. Radio Fuses.
7. Automatic Stabilization Control Panel.
8. Landing Light Switches.

COPILOT.

The copilot must be able to substitute for the pilot in any of his duties and must be as familiar with them as with his own. The helicopter may be flown from the copilot's position alone. The copilot can assist the pilot in making all inspections and checks and can operate auxiliary equipment such as lights, windshield wiper, radios, etc, as the pilot flies the helicopter. It will be convenient for the copilot to turn the rotor clutch pump switches on and off and to equalize the manifold pressures of the engines by means of the quadrant throttles while the pilot operates the twist-grip throttle after the clutches have been engaged. The arrangement of the controls in the pilot's compartment is such that in addition to flying the helicopter, the copilot will operate the following equipment:

1. Main Rotor Blade Folding Panel.
2. Tail Rotor Coning Switch.
3. Instrument Fuses.
4. Nose Door Lock Lever.
5. Instrument Power Switch.
6. Tail Wheel Lock Lever.
7. Emergency Hydraulic Shut-Off Valve and Hydraulic Pump Lever.
8. Landing Gear Emergency Valve Switch.
9. First Aid Kit.

CREW MEMBER.

In addition to assisting the pilot and copilot in accomplishing the cabin interior check, the arrangement of controls in the helicopter is such that a crew member in the cabin may have to operate the following equipment:

1. Forward or Aft Cabin Interphone.
2. Manual Cargo Release Handle.
3. Cargo Sling Stowage Line.
4. Pilot's Compartment Ladder.
5. First Aid Kit.
6. Check Heater Compartment Fire Warning Light.
7. Auxiliary Power Unit.
8. Cargo Hoist.

Also, in case of electrical failure:

9. Nose Door, Ramp, and Blade Folding Manual Override Controls.
10. Battery Bus and High Amperage Circuit Breaker Panels; Transformer Fuse Panel.

Cabin personnel should check with the pilot before operating any equipment during flight. When the auxiliary power unit is used for engine starting, it should not be turned off until the pilot signals that the generator switches are on and that the generators are operating. Personnel should not enter the aft fuselage section in flight, except with the pilot's permission, to position the stabilizer manual override lever. Cabin personnel engaged in cargo sling operation should always wear a safety harness, clipped to a cargo tie-down ring, when the center hatchway is open in flight.

SECTION IX
ALL WEATHER OPERATIONS

INTRODUCTION.

The function of this section is to provide information relative to operation under conditions of cold weather, desert procedure, night flying, etc. This section will not include description of equipment since that information is presented in Sections I and IV.

ICE AND RAIN.

The problem of major concern in the ice and rain operation of helicopters is the restriction to visibility, especially during landings and take-offs. This problem is of paramount importance when operating from other than an operational air base. The danger of breaking through ice is minimized by maintaining maximum rpm when resting on an unknown ice surface.

WARNING

Flights during icing conditions should not be made because these helicopters are not equipped with anti-icing and de-icing equipment.

EXTERIOR INSPECTION.

In accomplishing the exterior inspection, the main rotor system, particularly the main blade dampers, should be checked for freedom of movement. The dampers may be checked from the main gear box platform by rocking the main rotor blades fore and aft. Check engine induction cold air intake screens and pitot tubes to determine that they are free from ice.

CAUTION

Remove all ice accumulations prior to flight.

TAXIING.

Little difference will be noted between normal taxiing and taxiing on ice. Helicopters should not be taxied on an ice other than on the hard surfaced areas of an air base.

CAUTION

Use caution when taxiing on iced surfaces because of the minimum braking action available.

TAKE-OFF.

Take-off from runways or other prepared take-off areas that are covered with ice may be considered normal, except that the pilot must ascertain that the helicopter is completely free of the ice. A slight yawing movement, induced by light tail rotor pedal motion with throttle near maximum rpm, should be made to break the wheels free from any possible seizing.

DURING FLIGHT.

The radio compass is susceptible to static caused by precipitation. Therefore, when flights are made under these conditions, the pilot should expect poor radio range reception.

WARNING

During icing conditions, the main rotor assembly and blades will collect ice. After a sufficient amount has accumulated, vibration or feedback will be felt in the cyclic control stick. An unbalanced blade condition, with probable loss of control, will result when portions of ice are thrown from the rotor blade. A landing should be made as soon as ice is detected.

Note

The windshield wipers may not operate properly and forward visibility may become distorted or obscured by precipitation. However, limited visibility is possible through the side windows.

LANDING.

Landings on runways or other prepared landing areas that are covered with ice may be considered normal, except that a normal landing should be made with a minimum hover before touchdown.

CAUTION

The droop stops may not function properly during shutdown if the helicopter has been flown during icing conditions. In event of droop stop malfunction, the cyclic control stick should be held well forward of the neutral position to insure maximum tail cone clearance.

TURBULENCE AND THUNDERSTORMS.

Turbulent areas should be avoided, if possible. However, if inadvertent turbulence is encountered, the helicopter should be flown at the recommended instrument airspeeds to improve its flying qualities and decrease the chance of blade stall. If severe turbulence is encountered, reduce airspeed to 70 knots and land, if possible. In moderate turbulence, the work load of the pilot increases to the extent that he must concentrate solely on flying the helicopter. As a result, the pilot must disregard navigation, position reports, etc. Because of the helicopter's relatively slow speed and the difficulty of handling the helicopter in turbulence, thunderstorm flying is not recommended.

Note

If thunderstorms are encountered during flight, land and wait for the storm to pass. Moderate turbulence and restricted visibility can be expected.

NIGHT FLYING.

Night flying presents the same problem as instrument flying, plus additional problems induced by illumination of the pilot's compartment and instruments, and by reflections from exterior lights.

Note

- The rotating anti-collision light should be used on the ground, whenever necessary, to preclude a collision.
- Use of the anti-collision light on the ground should be kept to an absolute minimum to avoid the excessive heat created on the ground which is detrimental to bulb life and increases maintenance problems, and to prevent the possible confusion of rescue operations, since emergency ground vehicles use a similar light.

Interior lights should be adjusted as required to provide adequate or proper illumination.

TAKE-OFF.

There is basically little difference in the technique used on night take-off from that used in day operations. Care should be used to make a clean decisive break from the ground to a safe hovering altitude. The landing light should be used to illuminate the ground. Position the beam of the light to provide a good forward reference.

CAUTION

The landing light should be extended and positioned for immediate use in event of an emergency.

Note

- The rotating anti-collision light is used continuously during flight to preclude a collision.
- The rotating anti-collision light should be turned off during flight through actual instrument conditions. With the light on during instrument conditions, the pilot could experience vertigo as a result of the rotating reflections of the light against the clouds. In addition, the light would be ineffective as an anti-collision light during instrument conditions since it could not be observed by pilots of other aircraft.

APPROACH AND LANDING.

The landing light should be used for all landings. In poorly lighted or unlighted areas, the landing light can be used to clear the landing area prior to landing. The light can be preset for landing or adjusted throughout the final approach.

COLD WEATHER OPERATION.

The following operating instructions are presented in addition to instructions in Section II, and should be complied with when cold weather conditions are encountered.

CAUTION

When operating in cold weather conditions, ascertain that the helicopter has been serviced with the proper lubricants. Refer to the servicing diagram (figure 1—45).

The major problems in cold weather operation are engine starting, the restrictions to visibility as a result of blowing snow (from the main rotor wash) especially during take-off and landing, and the ever present possibility of breaking through surface or hidden crust during landing. When cold weather engine starts are anticipated, oil dilution must be accomplished during the engine shutdown procedure.

PREPARATION FOR FLIGHT.

During cold weather operations, engine preheating is recommended to assist in engine starts. With the combined use of the winterization kit and the auxiliary power plant, the engine starting problem is greatly simplified. The 600,000 BTU heater should be used to preheat the engines until the cylinder head temperature gages indicate approximately 0°C (32°F). It is recommended that the winterization kit be used when the ambient temperature is below —32°C (—25°F). Approximate times for engine preheat at various temperatures are as follows:

0°F (—18°C) – 10 minutes
—20°F (—29°C) – 20 minutes
—40°F (—40°C) – 30 minutes
—65°F (—54°C) – 40 minutes

Note

If a winterization kit is not installed and auxiliary ground equipment is used for preheating, the listed engine preheat times are not applicable. Under these conditions, continue to use preheat until the cylinder head temperature gages register approximately 0°C (32°F).

The following procedure may be used to preheat the engines: Cap off the left-hand side of the heating system plenum chamber and close the plenum chamber damper levers. Operate the heater on helicopter battery power, and by use of the right engine main preheating duct, preheat the APU for a maximum of three minutes. When cold weather engine starts are anticipated, consideration must also be given to starting the APU under low temperature conditions, as the battery is unable to sustain the cabin heater for a long enough period of time to properly preheat the APU. To facilitate starting, APU oil may be diluted. The oil capacity of the APU is three quarts. If MIL-L-8383 oil is used, dilute 10 percent (0.3 quarts = 10 percent) below —20°F. The APU may then be started by hand or by the helicopter battery to supply power for the operation of the heater to preheat the engines. When preheating the engines, use the engine preheat switch, located on the overhead switch panel in the pilot's compartment, to allow the heater to cycle at a higher temperature. For operation of the APU and of the heater, refer to Section IV. With the installation of protective covers to shroud each engine nacelle nose section, and by use of the two large flexible ducts (part of the winterization kit), the helicopter may be preheated in extreme low temperature regions without recourse to auxiliary ground equipment. The engines are preheated by fastening the covers over the nacelles, sealing off the engine cooling air entrance and exits and the oil cooler intake duct. The two large flexible hoses are attached to the engine preheat duct connections (8, figure 1—2) at each end of the plenum chamber. Access to the connections is gained by removing a panel on each side of the main gear box fairing. The hoses lead over the wings and are connected at the other end to ports cut into the outboard side of the nacelle covers. Both hoses may be connected to one engine cover for quicker preheating of one engine, if desired.

Note

Commence engine preheating when beginning the preflight check so that the engines will be warmed up sufficiently for engine starting at the completion of the preflight check.

In addition to accomplishing the normal exterior inspection, be sure that the fuel and oil tank sumps are drained of water. If there is no flow, preheat to insure some flow. Check the tires, rotor brake accumulator, and the main landing gear shock struts for proper pressure at low temperatures. Also, check that the engine oil pressure transmitter line has been purged for low temperature operation.

STARTING ENGINES.

Since engine preheat has been applied, the normal starting procedure should be used. If oil pressure does not register almost immediately, shut down the engine to prevent damage due to a lack of lubrication.

ENGINE GROUND OPERATION.

On cold days it may require too long a time to warm up the engine until the oil temperature reaches 40°C by following the regular procedure. Under these conditions, engine rpm may be increased above 1500 rpm until the oil temperature stabilizes and the engine operates satisfactorily. In no case should the throttle be advanced to the point where maximum oil pressure is exceeded.

CAUTION

- The engine oil pressure gages should be monitored frequently to assure proper engine lubrication.
- Check that the engine overboard breathers are free of icing during prolonged engine idle periods.

It is recommended that the rotor be engaged soon after starting as the load on the engines will be instrumental in avoiding icing of the engine and oil tank breather lines. After rotor clutch engagement, monitor the transmission oil pressure gage to assure proper lubrication of the main gear box.

TAXIING.

WARNING

In cold weather, make sure that all instruments have warmed up sufficiently to insure normal operation. Check for sluggish instruments during taxiing. Little difference will be noted between normal taxiing and taxiing on snow. Helicopters should not be taxied on snow other than on the hard surfaced areas of an air base. If visibility is restricted by blowing snow or if the power required to taxi is excessive, the helicopter should be towed to take-off position.

CAUTION

- Use caution when taxiing on surfaces with snow because of the minimum braking action available.
- Retract the landing light when taxiing in loose snow to preclude damage to the retraction mechanism.

TAKE-OFF.

CAUTION

Take-off is prohibited with snow or ice on the helicopter. Failure to remove all snow and ice accumulated on the helicopter, including all rotor blades, while on the ground, can result in serious aerodynamic and structural effects when flight is attempted. Depending on the weight and distribution of the snow and ice, take-off, hovering, and climb-out performances can be adversely affected. This roughness pattern and location of the snow and ice can affect stall speeds and handling characteristics to a dangerous degree. In-flight structural damage can also result due to vibrations induced in flight by unbalanced loads of unremoved accumulations. These hazards can be eliminated by removing all snow and ice accumulations prior to flight.

Snow take-offs may be considered normal except for the following precautions that should be observed: Select an area devoid of loose or powdery snow to minimize the restriction to visibility from blowing snow. Before attempting to take off, roll the helicopter forward to insure that all tires are free and not frozen to the surface.

DURING FLIGHT.

Extreme caution should be exercised during hoisting operations or cargo sling operations under extremely low temperatures because of the possibility of freezing personnel with the rotor blast.

CAUTION

Static electricity resulting from atmospheric conditions should be dissipated by allowing the hoist to touch the ground prior to being handled by personnel. Otherwise, those touching the ground and the hoist will be shocked by the electrical charge. A short length of weighted ground wire should be provided as necessary for attachment to the hoist sling to establish a ground for dissipation of static electricity during hoisting operations.

During flight, insure continued availability of cyclic and collective servo control and cabin heat. Continued flight with loss of either cabin heat or flight control servos should be made with extreme caution.

BEFORE LANDING.

A normal descent should be made; however, select a landing area, if possible, devoid of loose powdery snow so that visibility will not be restricted by blowing snow.

LANDING.

Loose powdery snow and crusts (surface and hidden) should be anticipated on all landings on snow. Accomplish an approach with minimum hover before touchdown to minimize the rotor wash on loose powdery snow. Limited visibility will result if hovering is attempted before touchdown. If possible, landing should always be made where visual ground reference can be maintained. The reference point should be kept immediately forward and to the right so that it will be visible to the pilot at all times.

STOPPING OF ENGINES.

OIL DILUTION.

Prior to engine shutdown, the engine oil system should be diluted with fuel to retain oil fluidity at low temperatures. Before diluting, check the oil level and drain or fill as needed to 13.3 gallons.

1. Idle engine at 1000 rpm until oil temperature falls to 40°C (104°F).

Note

- Check that the engine overboard breathers are free of icing during prolonged engine idle periods.
- Dilution of the engine oil should not be accomplished with the oil temperature above 40°C, since the heat of the oil will evaporate some of the fuel, resulting in improper dilution.

2. Oil dilution manual shut-off valve handle – OPEN.
3. Fuel booster pump switch – NORM.

OIL DILUTION TIME

GRADE OIL	AIR TEMP °C	AIR TEMP °F	TIME Minutes	DILUTION Percent
1100	–7	20	1.0	5
	–12	10	1.7	10
	–20	–5	4.5	15
	–29	–20	6.0	20
	–37	–35	8.25	25
	–46	–50	10.5	30
1065	–21	–5	1.7	5
	–29	–20	*	10
	–37	–35	*	15
	–46	–50	6.0	20

Note

1. Dilution of grade 1100 oil will be conducted only when grade 1065 and/or preheat is not available and continuous temperatures below 25°F are expected.
2. Dilution of grade 1065 oil will be conducted only when preheat is not available and continuous temperatures below 10°F are forecast.

Figure 9–1. Oil Dilution Time

4. Oil dilution switch – Hold in the ON position. Refer to oil dilution time (figure 9–1).

Note

A considerable drop in fuel pressure is an indication that the oil dilution system is functioning properly.

5. Bleed oil pressure transmitter lines.
6. At the end of the dilution period, accomplish a normal engine shutdown but hold the oil dilution switch on until the engine stops.

Note

Check the oil supply, after a thorough warm-up, at the next starting of an engine in which the oil has been diluted.

BEFORE LEAVING THE HELICOPTER.

After the engines have stopped, protective covers should be installed. The fuel and the oil sumps should be drained of water. If the helicopter is to remain unsheltered at temperatures below –20°F, the battery should be removed to a heated area. Check that the battery is reinstalled before the next engine start.

HOT WEATHER OPERATION.

More power will be required to hover during hot weather than on a standard day. Hovering ceilings will be lower for the same gross weight and power settings on a hot day. High rpm should be maintained at all times to insure sufficient power and control, especially during take-offs and hovering. Figure A—20 shows maximum gross weights at which the helicopter can be hovered out of ground effect with various combinations of altitude, temperature, specific humidity, and headwind.

DESERT OPERATION.

The major problem in desert operations is the restriction to visibility as a result of the dust cloud created by the rotor downwash. Consequently, take-offs, cargo sling operations, and landings in sandy or dusty areas must, at times, be made without reference to the horizon and surrounding objects. This restriction to visibility can be overcome by proper use of the pilot's landing light. Operation in dusty or sandy areas can also be harmful to the engines. Installation of air filters will help to minimize damage to the engines.

BEFORE ENTERING THE HELICOPTER.

The helicopter should be towed into take-off position, which, if at all possible, should be on a hard clean surface, free from sand. After the helicopter is towed into position, accomplish the normal exterior inspection. Refer to Section II. Perform the following, in addition to the requirements of figure 2—1:

1. Oleo struts — Check for presence of sand and grit. Clean in accordance with approved methods, if necessary.
2. Rotor blades — Check for evidence of sand abrasion.
3. Protective covers — Removed.
4. Fuselage and engine nacelles — Check for sand that has filtered through openings.
5. Engine and main gear box oil coolers — Check for sand clogging and contamination.

ON ENTERING THE HELICOPTER.

Open all vents, windows, and doors to increase ventilation. Perform the normal interior inspection. Refer to Section II.

ENGINE STARTING AND GROUND OPERATION.

CAUTION

Do not start the engine during a dust or sand storm unless absolutely necessary, to avoid the danger of drawing sand into the carburetor induction air intakes resulting in premature wear of the engines.

If possible, engine starting and ground operation should be accomplished from a hard clean surface. This is extremely important after rotor clutch engagement as the downwash from the main rotor will stir up clouds of sand. Accomplish the normal engine start as presented in Section II. Limit ground operation of the engines to a minimum. Every effort should be made to minimize blowing the sand up around the main rotor assembly and the engine nacelles by utilizing minimum pitch and engine rpm.

TAKE-OFF.

Increase engine to maximum rpm. If the rotor should stir up dust and sand, it would be best to increase collective pitch and take off; do not hover in ground effect but continue to climb rapidly. Refer to **TAKE-OFF (NORMAL VERTICAL)**, Section II. Take-offs, unless necessary, should be avoided during a dust or sand storm.

DURING FLIGHT.

When sand storm conditions are encountered, evasive action should be taken to prevent extensive damage to the helicopter, particularly to the rotor blades and engines. If the storm area is not too extensive, it may be possible to fly over or around it, but never through it. If practicable, return to the base so that the helicopter may be tied down and all protective covers installed. If this is not possible, land and shut down. Head the helicopter into the wind, and, if possible, tie down the blades. Close all doors, windows, and vents and remain in the helicopter. When the storm is over, perform an exterior inspection (figure 2—1) before resuming flight. Clean out as much as possible of the sand that has sifted into the fuselage and the engine nacelles, particularly the induction air and oil cooling systems. Check the rotor blades for damage and the flight control system for excessive friction.

CARGO SLING LOAD OPERATIONS.

When a cargo sling load hook-up is to be made in a dusty or sandy area, make an approach at a slightly steeper angle than in the normal approach, terminating the approach slightly higher than the upper limit of ground cushion. This enables the pilot to orient over the load and start a vertical descent through the dust cloud created by the rotor downwash as the helicopter descends into ground effect. The dust cloud is initially blown out and away from the helicopter, and ground visibility can be maintained by looking straight down until hovering height is reached. As the dust cloud builds, obstructing visibility, adjust the landing light down and to the right. This creates a "ball of light" on the ground from which the height and position of the aircraft can be determined. Determine the approximate altitude desired for hook-up operations and adjust the landing light at a hover prior to entering the hook-up area. While the pilot's attention is centered on the "ball of light," the copilot will monitor all instruments and override the pilot on the twist-grip throttle to maintain the proper engine rpm. A light placed on the ground, out in front of the helicopter, can be seen through the thin upper portion of the dust cloud as the load is brought to a hover, and establishes a point of reference for the take-off out of the dust cloud.

APPROACH AND LANDING.

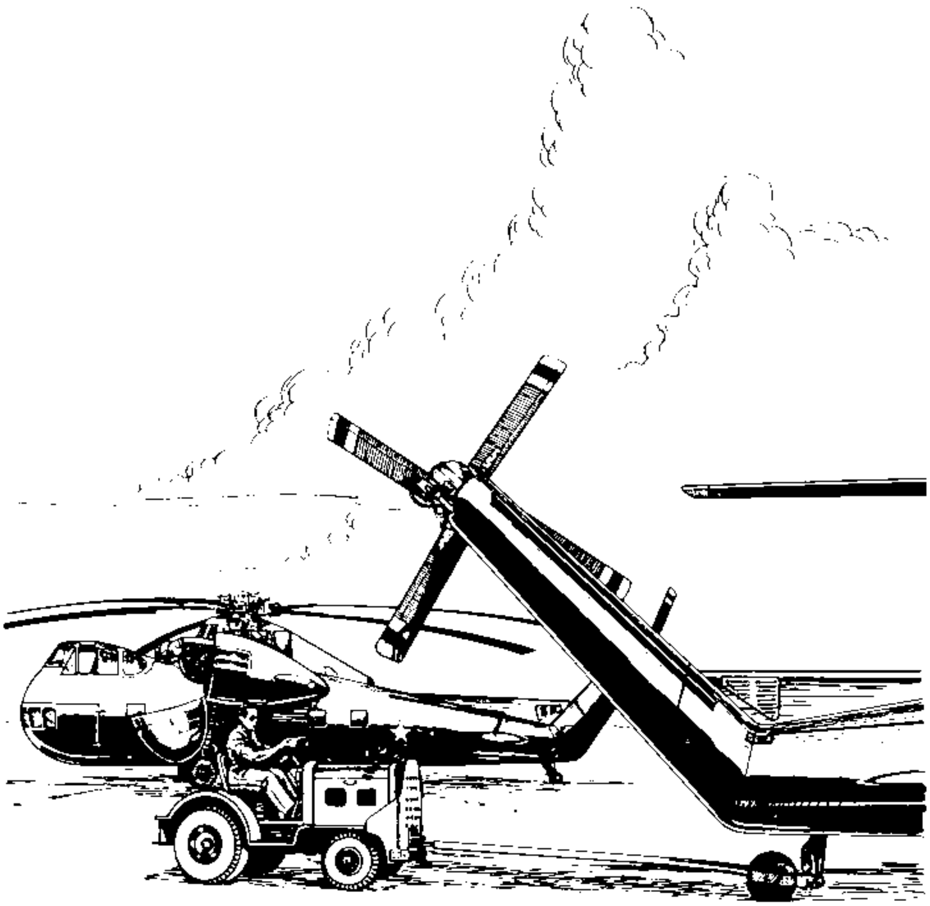
The landing approach should be relatively steep at low airspeed, with little forward speed just above the surface, and landing without hovering. Try to land on a hard surface or in an area where sand will not be blown on other helicopters. Restrict taxiing to a minimum.

POSTFLIGHT ENGINE CHECK.

Accomplish POSTFLIGHT ENGINE CHECK, Section II as soon as possible.

BEFORE LEAVING THE HELICOPTER.

Close all doors, windows, and vents. Be sure that all protective covers are installed. Refer to **BEFORE LEAVING THE HELICOPTER,** Section II.



APPENDIX I
OPERATING DATA

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont)

INTRODUCTION.

The purpose of the charts in Appendix I is to aid the pilot in selecting proper power settings, airspeeds, and altitudes to be used to obtain optimum performance from the helicopter. Figures A–10 through A–21 are applicable to helicopters with 72-foot main rotors, both 14°, 21.5 inch chord blades and 8°, 23.65 inch chord blades. However, helicopters with 72-foot main rotor 14°, 21.5 inch chord blades are restricted to 80 knots maximum airspeed in order to increase service life of the blades.

APPARENT VS ACTUAL ROTOR RPM.

The table (figure A–1) shows the relationship between apparent rotor rpm, as indicated on the rotor pointer of the engine rotor tachometer, and actual rotor rpm.

AIRSPEED INSTALLATION CORRECTIONS.

Airspeed installation corrections to convert indicated airspeed to calibrated airspeed are shown in figure A–3.

CAUTION

To avoid erroneous indication of the airspeed indicator, keep the pilot's sliding window closed during flight. The open window creates turbulence or a low pressure area at the entrance of the pitot tubes resulting in false airspeed indications.

ENGINE OPERATING LIMITS CURVES.

On the engine operating limits curves (figure A–4), the constant rpm lines sloping downward as altitude is increased, represent full throttle operation. The short lines crossing the rpm lines join the points of equal manifold pressure. The horizontal portions of the rpm lines at the left side of the charts indicate horsepower limitations at partial throttle settings. The line joining the points where the rpm lines change from horizontal to sloping, represents the critical altitude. When operating below critical altitude, the manifold pressure must be watched to prevent exceeding engine limitations at any selected rpm. Above the critical altitude, power drops off even though full throttle is used.

TAKE-OFF DISTANCES.

Take-off distances charts (figures A–10, A–11, and

APPARENT VS ACTUAL ROTOR RPM (Ratio of 14.01 to 1)	
APPARENT ROTOR RPM	ACTUAL ROTOR RPM
1000	71
1100	78
1200	86
1300	93
1400	100
1500	107
1600	114
1700	121
1800	129
1900	136
2000	143
2100	150
2200	157
2300	164
2400	171
2500	178
2600	186
2700*	193
2800	200
3020	215**
*ENGINE RED LINE	**ROTOR RED LINE

Figure A–1. Apparent Vs Actual Rotor RPM

A–12) show the IAS value required before beginning to climb, the accelerating distance to attain this airspeed, and the total distance to clear a 50-foot obstruction. The distances are computed for various gross weight, altitude, and temperature conditions. Vertical take-offs can be made under conditions where zeros appear on the chart; however, caution should be observed when operating in the red area shown in figure 5–11. Unmarked accelerating distances are for vertical take-off with air acceleration at 10-foot wheel clearance. Acceleration distances marked (*) are for ground run on a prepared surface runway. Take-off data is based on take-off power.

CLIMB CHART FOR NORMAL POWER.

The climb charts (figures A–13, A–14, and A–15) indicate the maximum rate-of-climb at best rate-of-climb

Figure A–2 deleted.

speed. Performance is computed at various gross weight, temperature, and altitude conditions with the time to climb and the fuel consumed in climbing from sea level to altitude. The fuel consumed includes a warm-up and take-off allowance of 200 pounds which is equivalent to the fuel consumed for five minutes operation at Normal Power. Climbs are computed for Normal Power.

LANDING DISTANCE CHARTS.

Landing distance charts (figure A–16) include the best power off IAS approach speed, ground roll, and the corresponding distance to clear a 50-foot obstruction at various gross weight, altitude, and temperature conditions. With gross weight, altitude, and temperature conditions where hovering out of ground effect is possible, vertical descents can be made.

FLIGHT OPERATION INSTRUCTION CHARTS.

Each chart is divided into two sections, Range and Endurance. Column I shows the range at various altitudes based on maximum continuous cruise power settings. Column II is based on medium cruise power setting, and Column III is based on maximum air range power settings. The flight operation instruction charts are based on flight test. Column A indicates maximum endurance for minimum speed with the corresponding power settings and fuel consumption at various altitudes. Column B indicates the power settings, airspeeds, and fuel consumption for maximum endurance, except as restricted by recommended confortable cruise speeds. Refer to

AIRSPEED INSTALLATION CORRECTIONS 72-FOOT MAIN ROTOR						
	LEVEL FLIGHT		FORWARD CLIMB		AUTOROTATION	
INDICATED AIRSPEED Knots	AIRSPEED CORRECTION Knots	CALIBRATED AIRSPEED Knots	AIRSPEED CORRECTION Knots	CALIBRATED AIRSPEED Knots	AIRSPEED CORRECTION Knots	CALIBRATED AIRSPEED Knots
30	+5	35				
40	+5	45			+16	56
50	+4	54	−27	23	+13	63
60	+4	64	−18	42	+11	71
70	+4	74	−17	53	+ 9	79
80	+3	83	−14	66	+ 6	86
90	+3	93			+ 4	94
100	+3	103				
110	+3	113				
120	+4	124				
130	+5	135				
Based on flight test data as of December 1, 1958						

Figure A—3. Airspeed Installation Corrections — 72-Foot Main Rotor

figures A—17, A—18, and A—19. For any planned extended flights using auxiliary tanks, oil consumption should be considered.

USE OF CHARTS.

To clarify the use of the various charts, the following example problem is provided:

It is necessary for a helicopter with a 72-foot main rotor (8°, 23.65 inch chord blades), clean configuration, to fly 4000 pounds of cargo to a point 50 nautical miles from the point of take off, land, unload, and then pick up 20 men and return them to the original take-off point. The take-off and landing points are at sea level, but because of the terrain, it is necessary to accomplish the flight at 2000 feet. Temperature is 15°C at sea level. Gross weight at take-off is 28000 pounds, including 2400 pounds of fuel. Determine fuel required and cruising speeds to be flown.

1. Fuel required for warm-up, take-off, and climb to 2000. (From figure A—13 at 28500 pounds and 15°C.) ..... 266 pounds
2. Fuel required for cruising outbound 50 nautical miles at 2000 feet altitude. (From figure A—17, sheet 2, bottom of column III.)

   $\frac{\text{(Distance)}}{\text{TAS}} \times \text{(LBS/HR)} = \frac{50}{105}$
   $\times$ 1390 = ... 662 pounds

   Power settings: 36.2 inches Hg and 2500 rpm with normal mixture. True airspeed 105 knots.
3. Fuel required for letdown from 2000 feet and landing (assumed to be 0 pounds with 0 distance covered). ..... 0 pounds
4. Total fuel consumed on outboard leg. 928 pounds
5. Fuel required for warm-up, take-off, and climb to 2000 feet at new gross weight of 26472 pounds.
   28000 — (fuel consumed) 928 — (cargo) 4000 + (20 men) 3400 = 26472 pounds (From figure A—13 at 28500 pounds and 15°C.) ..... 266 pounds
6. Fuel required for cruising 50 nautical miles on return leg at 2000 feet. (From figure A—17, sheet 2, column III.) ..... 662 pounds
   Power settings: 36.2 inches Hg and 2500 rpm normal mixture. True airspeed 105 knots.

7. Fuel required for let down from 2000 feet and landing (assumed to be 0 pounds with 0 distance covered). ...... 0 pounds
8. Total fuel consumed on inbound leg. 928 pounds
9. Total fuel consumed on flight. .... 1856 pounds
10. Reserve fuel = initial fuel (2400 pounds) — fuel consumed (1856 pounds = 544 pounds or 27 percent.

MAXIMUM GROSS WEIGHT FOR HOVERING OUT OF GROUND EFFECT.

The maximum gross weights at which the helicopter can be hovered out of ground effect, with take-off power, at various combinations of altitude, temperature, specific humidity, and headwinds may be determined from this chart (figure A—20).

USE OF THE HOVERING OUT OF GROUND EFFECT CHART.

Determine the maximum gross weight to hover out of ground effect using take-off power at 6400 feet pressure altitude with an operating air temperature of 15°C (59°F), a specific humidity of 0.01 and a headwind of 15 knots.

1. Enter the chart at a pressure altitude of 6400 feet (Point A).
2. From Point A move horizontally to an OAT of 15°C (59°F) (Point B).
3. From Point B move vertically to the specific humidity base line (Point C).
4. From Point C move parallel to the specific humidity influence lines to a specific humidity of 0.01 (Point D).
5. From Point D move vertically to the headwind base line (Point E).
6. From Point E move parallel to the headwind influence lines to a headwind of 15 knots (Point F).
7. From Point F move vertically to the gross weight (Point G). The maximum gross weight to hover under the above conditions would be 27700 pounds.

MAXIMUM GROSS WEIGHT FOR HOVERING IN GROUND EFFECT.

The maximum gross weights at which the helicopter can be hovered in ground effect (10-foot wheel clearance) with take-off power, at various combinations of altitude, temperature, specific humidity, and headwinds may be determined from this chart (figure A—21).

USE OF THE HOVERING IN GROUND EFFECT CHART.

Determine the maximum gross weight to hover with a 10-foot wheel clearance using take-off power at 8600 feet pressure altitude with an operating air temperature of 35°C (95°F), a specific humidity of 0.02 and a 20-knot headwind.

1. Enter the chart at a pressure altitude of 8600 feet (Point A).
2. From Point A move horizontally to an OAT of 35°C (95°F) (Point B).
3. From Point B move vertically to the specific humidity base line (Point C).
4. From Point C move parallel to the specific humidity influence lines to a specific humidity of 0.02 (Point D).
5. From Point D move vertically to the headwind base line (Point E).
6. From Point E move parallel to the headwind influence lines to a headwind of 20 knots (Point F).
7. From Point F move vertically to the gross weight (Point G). The maximum gross weight to hover under the above conditions would be 26280 pounds.

ENGINE OPERATING LIMITS

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

ENGINES:(2) R-2800-54
FUEL GRADE: 115/145

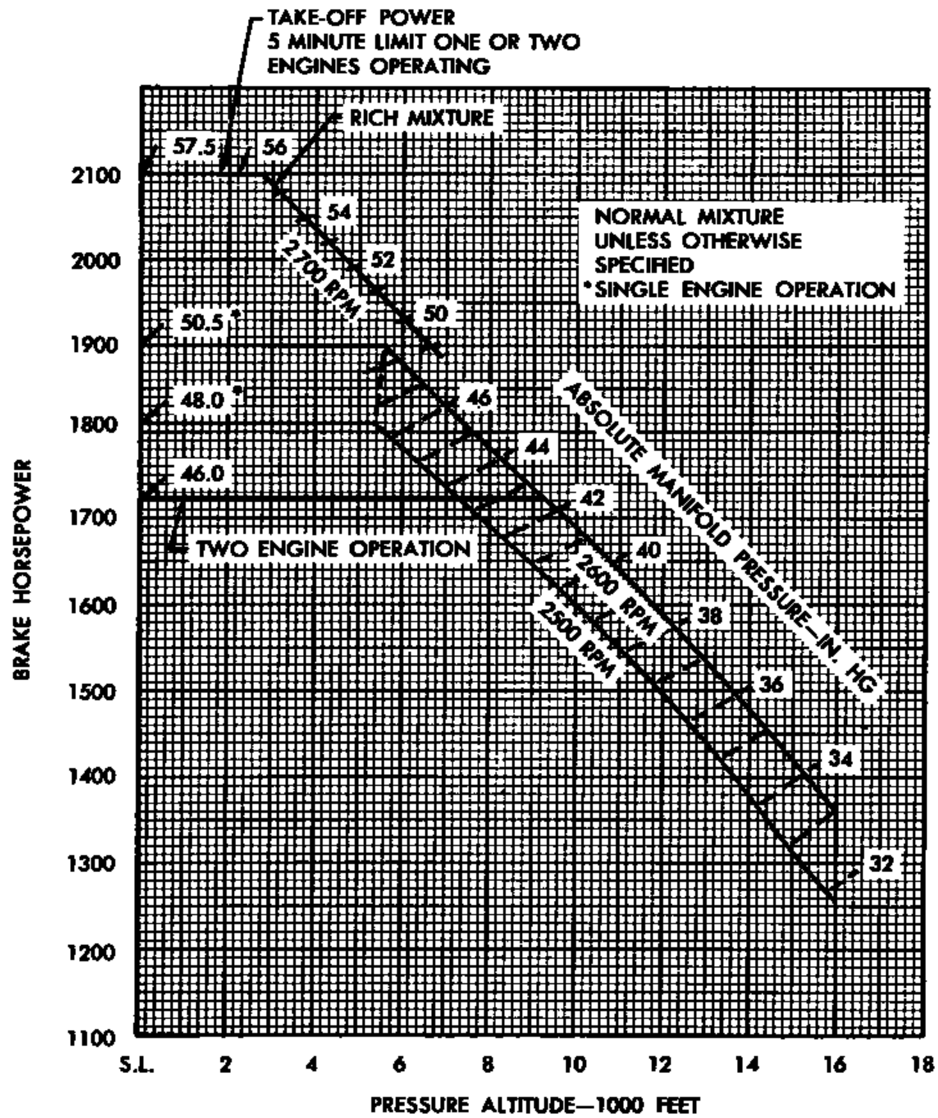


Figure A—4. Engine Operating Limits

Pages 185 through 190, figures A—5, A—6, A—7, A—8 (sheets 1 & 2), and figure A—9 deleted.

TAKE-OFF DISTANCES – FEET

FIRM DRY SOD
CONFIGURATION: CLEAN
72 FOOT MAIN ROTOR

MODEL: HR2S-1 ENGINES: (2) R-2800-54

GROSS WEIGHT LB	PRESSURE ALTITUDE 1000 FEET	−25 DEG CENT.			−5 DEG CENT.			+15 DEG CENT.			+35 DEG CENT.			+55 DEG CENT.		
		IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′
31,000	SL	0	0	0	0	0	0	7	45	253	15	167	454	19	357	736
	2	0	0	0	0	0	0	13	138	568	17	276	644	24	696	1144
	4	0	0	0	9	92	357	19	368	696	25	851	1369	34*	506*	1029*
	6	16	184	437	21	437	805	28*	316*	782*	35*	535*	1133*	41*	868*	1800*
	8	30	1035	1518	35*	460*	1064*	45*	828*	1725*						
	10	46*	805*	1903*												
	12															
28,500	SL	0	0	0	0	0	0	0	0	0	0	0	0	6	40	253
	2	0	0	0	0	0	0	0	0	0	6	35	207	15	190	460
	4	0	0	0	0	0	0	7	46	253	15	219	535	24	621	1000
	6	0	0	0	9	81	299	17	276	638	24	719	1173	32*	437*	949*
	8	15	161	380	19	345	713	24	874	1380	34*	506*	1070*	36*	661*	1633*
	10	25	725	1271	35*	454*	989*	38*	661*	1587*						
	12	46*	828*	1765*												
26,000	SL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	4	0	0	0	0	0	0	0	0	0	0	0	0	14	155	380
	6	0	0	0	0	0	0	6	34	190	14	184	443	20	414	805
	8	0	0	0	6	46	230	15	207	506	24	658	1000	28*	328*	828*
	10	15	161	328	15	236	598	24	736	1173	31*	437*	972*	35*	621*	1374*
	12	25	581	1006	27	1035	1679	35*	552*	1334*	45*	1024*	2530*			
23,500	SL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	6	0	0	0	0	0	0	0	0	0	0	0	0	6	46	247
	8	0	0	0	0	0	0	0	0	0	11	127	339	15	242	598
	10	0	0	0	0	0	0	15	196	414	18	374	771	24	920	1380
	12	6	35	207	14	196	483	22	575	966	25*	288*	845*	34*	598*	1219*

REMARKS:
1. DISTANCES BASED ON 4200 BHP AND 2700 RPM AT SEA LEVEL
2. ACCELERATION DISTANCES UNMARKED ARE FOR VERTICAL TAKE-OFF AND AIR ACCELERATION
3. ACCELERATION DISTANCES MARKED (*) ARE FOR GROUND RUN—PREPARED SURFACE REQUIRED

BASED ON: ESTIMATED DATA
DATA AS OF: DECEMBER 1, 1958

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure A–10. Take-Off Distances – 72-Foot Main Rotor

TAKE-OFF DISTANCES – FEET

FIRM DRY SOD

CONFIGURATION: ONE 150 OR 300 GAL AUXILIARY TANK

MODEL: HR2S-1 72 FOOT MAIN ROTOR ENGINES: (2) R-2800-54

GROSS WEIGHT LB	PRESSURE ALTITUDE 1000 FEET	−25 DEG CENT.			−5 DEG CENT.			+15 DEG CENT.			+35 DEG CENT.			+55 DEG CENT.		
		IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′	IAS KNOTS	ACCEL DIST	CLEAR 50′
31,000	SL	0	0	0	0	0	0	7	46	253	14	178	458	20	385	736
	2	0	0	0	0	0	0	13	158	368	17	288	658	24	713	1150
	4	0	0	0	12	121	362	19	368	713	24	851	1374	35*	506*	1035*
	6	16	190	437	22	460	828	29*	316*	794*	35*	535*	1150*	44*	914*	1932*
	8	30	1035	1558	35*	466*	1087*	46*	886*	1794*						
	10	46*	817*	1978*												
	12															
28,500	SL	0	0	0	0	0	0	0	0	0	0	0	0	7	46	265
	2	0	0	0	0	0	0	0	0	0	6	46	213	15	196	460
	4	0	0	0	0	0	0	7	46	270	15	230	550	25	633	1018
	6	0	0	0	10	92	305	17	276	638	24	736	1173	33*	454*	955*
	8	14	161	391	18	345	736	24	874	1397	34*	512*	1081*	36*	684*	1679*
	10	25	736	1305	35*	472*	1012*	39*	690*	1633*						
	12	46*	828*	1840*												
26,000	SL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	4	0	0	0	0	0	0	0	0	0	0	0	0	13	144	385
	6	0	0	0	0	0	0	6	35	190	15	190	449	20	426	811
	8	0	0	0	6	46	236	15	207	512	24	638	1012	28*	339*	834*
	10	15	161	339	16	236	598	24	748	1175	31*	437*	989*	35*	633*	1392*
	12	25	581	1018	28	1064	1702	34*	552*	1369*	44*	1050*	2668*			
23,500	SL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	6	0	0	0	0	0	0	0	0	0	0	0	0	7	52	253
	8	0	0	0	0	0	0	0	0	0	11	127	345	15	253	604
	10	0	0	0	0	0	0	15	196	426	18	385	776	24	932	1397
	12	6	40	207	15	207	483	22	575	972	25*	288*	857*	34*	592*	1242*

REMARKS: 1. DISTANCES BASED ON 4200 BHP AND 2700 RPM AT SEA LEVEL
2. ACCELERATION DISTANCES UNMARKED ARE FOR VERTICAL TAKE-OFF AND AIR ACCELERATION
3. ACCELERATION DISTANCES MARKED (*) ARE FOR GROUND RUN—PREPARED SURFACE REQUIRED

BASED ON: ESTIMATED DATA
DATA AS OF: DECEMBER 1, 1958

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure A–11. Take-Off Distances – 72-Foot Main Rotor

TAKE-OFF DISTANCES – FEET

FIRM DRY SOD
CONFIGURATION: TWO 150 OR 300 GAL
AUXILIARY TANKS

MODEL: HR2S-1 72 FOOT MAIN ROTOR ENGINES: (2) R-2800-54

GROSS WEIGHT LB	PRESSURE ALTITUDE 1000 FEET	−25 DEG CENT.			−5 DEG CENT.			+15 DEG CENT.			+35 DEG CENT.			+55 DEG CENT.		
		IAS KNOTS	ACCEL DIST	CLEAR 50'	IAS KNOTS	ACCEL DIST	CLEAR 50'	IAS KNOTS	ACCEL DIST	CLEAR 50'	IAS KNOTS	ACCEL DIST	CLEAR 50'	IAS KNOTS	ACCEL DIST	CLEAR 50'
31,000	SL	0	0	0	0	0	0	7	46	253	15	184	460	20	391	759
	2	0	0	0	0	0	0	14	161	391	18	299	667	24	713	1173
	4	0	0	0	13	150	368	21	414	771	24	851	1380	35*	523*	1041*
	6	15	195	460	22	472	851	30*	345*	811*	35*	535*	1156*	44*	920*	1978*
	8	31	1087	1604	35*	472*	1110*	46*	886*	1846*						
	10	46*	828*	2095*												
	12															
28,500	SL	0	0	0	0	0	0	0	0	0	0	0	0	8	52	270
	2	0	0	0	0	0	0	0	0	0	6	46	230	15	196	472
	4	0	0	0	0	0	0	7	46	276	15	230	558	24	633	1029
	6	0	0	0	9	92	316	17	288	644	24	748	1196	33*	460*	978*
	8	15	161	403	18	345	753	24	891	1409	35*	529*	1093*	36*	679*	1731*
	10	26	765	1328	35*	472*	1035*	39*	690*	1714*						
	12	46*	840*	1926*												
26,000	SL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	4	0	0	0	0	0	0	0	0	0	0	0	0	13	144	391
	6	0	0	0	0	0	0	6	40	196	15	196	460	20	437	840
	8	0	0	0	6	46	236	15	207	523	24	638	1012	28*	328*	851*
	10	14	161	339	15	236	610	24	748	1196	31*	437*	1012*	35*	633*	1426*
	12	25	581	1024	28	1070	1748	35*	564*	1415*	46*	1075*	2921*			
23,500	SL	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	6	0	0	0	0	0	0	0	0	0	0	0	0	7	46	265
	8	0	0	0	0	0	0	0	0	0	11	121	345	20	368	610
	10	0	0	0	0	0	0	15	207	437	18	391	782	24	937	1415
	12	6	46	219	14	201	495	23	621	983	26*	316*	868*	34*	598*	1265*

REMARKS: 1. DISTANCES BASED ON 4200 BHP AND 2700 RPM AT SEA LEVEL
2. ACCELERATION DISTANCES UNMARKED ARE FOR VERTICAL TAKE-OFF AND AIR ACCELERATION
3. ACCELERATION DISTANCES MARKED (*) ARE FOR GROUND RUN—PREPARED SURFACE REQUIRED

BASED ON: ESTIMATED DATA
DATA AS OF: DECEMBER 1, 1958

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure A—12. Take-Off Distances — 72-Foot Main Rotor

CLIMB CHART FOR NORMAL POWER

CONFIGURATION: ONE 150 OR 300 GAL. AUXILIARY TANK: GEAR UP

MODEL: HR25-1 72 FOOT MAIN ROTOR ENGINES: (2) R-2800-54

TEMPERATURE DEGREES CENTIGRADE	PRESSURE ALTITUDE FEET	GROSS WEIGHT LB															
		31,000				28,500				26,000				23,500			
		BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB	BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB	BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB	BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB
	SL	76	1210	0	200	76	1475	0	200	76	1764	0	200	76	2079	0	200
	2,000	75	1157	1.7	268	75	1432	1.4	255	75	1730	1.2	246	75	2054	1.0	239
	4,000	73	1091	3.5	339	73	1377	2.8	312	73	1685	2.3	293	73	2019	2.0	278
—25 DEGREES	6,000	71	1010	5.4	415	71	1309	4.3	372	71	1629	3.5	341	71	1974	3.0	318
	8,000	69	705	7.6	505	69	1226	5.9	435	69	1559	4.8	391	69	1916	4.0	359
	10,000	67	290	11.5	662	67	907	7.7	508	67	1459	6.1	444	67	1829	5.0	402
	12,000									65	990	7.7	503	65	1540	6.2	646
	SL	76	1074	0	200	76	1341	0	200	76	1630	0	200	76	1946	0	200
	2,000	75	1013	1.9	277	75	1291	1.5	261	75	1590	1.2	250	75	1916	1.0	242
	4,000	73	938	4.0	359	73	1228	3.1	324	73	1539	2.5	301	73	1875	2.1	284
—5 DEGREES	6,000	71	847	6.2	448	71	1151	4.8	392	71	1475	3.9	354	71	1822	3.2	327
	8,000	69	492	9.1	566	69	1057	6.6	464	69	1396	5.2	410	69	1757	4.3	372
	10,000									67	1216	6.7	469	67	1588	5.5	418
	12,000													65	1293	6.9	467
	SL	76	944	0	200	76	1213	0	200	76	1504	0	200	76	1820	0	200
	2,000	75	875	2.2	288	75	1156	1.7	268	75	1458	1.4	254	75	1784	1.1	244
	4,000	73	790	4.6	384	73	1085	3.5	339	73	1399	2.8	310	73	1738	2.3	290
+15 DEGREES	6,000	71	630	7.3	492	71	999	5.4	416	71	1327	4.2	369	71	1679	3.4	337
	8,000									69	1239	5.8	431	69	1606	4.6	386
	10,000													67	1364	6.0	437
	12,000																
	SL	76	819	0	200	76	1090	0	200	76	1383	0	200	76	1700	0	200
	2,000	75	741	2.6	303	75	1026	1.9	276	75	1330	1.5	259	75	1659	1.2	248
	4,000	73	647	5.4	418	73	947	3.9	357	73	1265	3.0	320	73	1607	2.4	296
+35 DEGREES	6,000					71	851	6.1	446	71	1185	4.6	386	71	1541	3.7	347
	8,000													69	1444	5.0	400
	10,000																
	12,000																
	SL	76	696	0	200	76	972	0	200	76	1266	0	200	76	1586	0	200
	2,000	75	610	3.1	322	75	900	2.1	285	75	1208	1.6	265	75	1539	1.3	251
	4,000					73	812	4.5	379	73	1135	3.3	333	73	1481	2.6	304
+55 DEGREES	6,000									71	1047	5.2	407	71	1408	4.0	360
	8,000																
	10,000																
	12,000																

REMARKS: 1. CLIMBS BASED ON 3450 BHP AND 2600 RPM AT SEA LEVEL.
2. FUEL ALLOWANCE FOR WARM-UP AND TAKE-OFF, 200 LB (5 MIN AT 3450 BHP).

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure A–13. Climb Chart For Normal Power — 72-Foot Main Rotor

CLIMB CHART FOR NORMAL POWER

CONFIGURATION: CLEAN, GEAR UP.

MODEL: HR2S-1 | 72 FOOT MAIN ROTOR | ENGINES: (2) R-2800-54

TEMPERATURE DEGREES CENTIGRADE	PRESSURE ALTITUDE FEET	GROSS WEIGHT LB 31,000 BEST IAS KNOTS	31,000 RATE OF CLIMB FPM	31,000 FROM SL TIME MIN	31,000 FROM SL FUEL USED LB	28,500 BEST IAS KNOTS	28,500 RATE OF CLIMB FPM	28,500 FROM SL TIME MIN	28,500 FROM SL FUEL USED LB	26,000 BEST IAS KNOTS	26,000 RATE OF CLIMB FPM	26,000 FROM SL TIME MIN	26,000 FROM SL FUEL USED LB	23,500 BEST IAS KNOTS	23,500 RATE OF CLIMB FPM	23,500 FROM SL TIME MIN	23,500 FROM SL FUEL USED LB
25 DEGREES	SL	76	1231	0	200	76	1497	0	200	76	1787	0	200	76	2104	0	200
	2,000	75	1176	1.7	266	75	1453	1.4	254	75	1752	1.1	245	75	2077	.9	238
	4,000	73	1109	3.4	337	73	1397	2.8	310	73	1706	2.3	292	73	2041	1.9	277
	6,000	71	1028	5.3	412	71	1328	4.2	369	71	1648	3.5	339	71	1994	2.9	317
	8,000	69	726	7.5	500	69	1244	5.8	431	69	1578	4.7	389	69	1935	3.9	358
	10,000	67	310	11.3	650	67	928	7.6	503	67	1477	6.0	441	67	1848	5.0	400
	12,000									65	1011	7.6	499	65	1558	6.2	443
−5 DEGREES	SL	76	1096	0	200	76	1364	0	200	76	1655	0	200	76	1972	0	200
	2,000	75	1034	1.9	275	75	1313	1.5	260	75	1614	1.2	249	75	1940	1.0	241
	4,000	73	958	3.9	356	73	1249	3.1	322	73	1561	2.5	299	73	1898	2.1	282
	6,000	71	865	6.1	444	71	1170	4.7	388	71	1495	3.8	352	71	1844	3.1	325
	8,000	69	513	8.9	558	69	1076	6.5	459	69	1415	5.2	406	69	1778	4.2	369
	10,000									67	1234	6.6	465	67	1608	5.4	415
	12,000													65	1312	6.8	464
+15 DEGREES	SL	76	967	0	200	76	1238	0	200	76	1530	0	200	76	1847	0	200
	2,000	75	897	2.1	286	75	1179	1.6	266	75	1482	1.3	253	75	1810	1.1	244
	4,000	73	811	4.5	380	73	1107	3.4	336	73	1422	2.7	308	73	1762	2.2	288
	6,000	71	654	7.1	486	71	1019	5.3	411	71	1349	4.2	366	71	1701	3.4	334
	8,000									69	1260	5.7	427	69	1627	4.6	382
	10,000													67	1385	5.9	433
	12,000																
+35 DEGREES	SL	76	812	0	200	76	1116	0	200	76	1410	0	200	76	1729	0	200
	2,000	75	763	2.5	300	75	1050	1.9	274	75	1356	1.5	258	75	1686	1.2	247
	4,000	73	668	5.3	411	73	969	3.8	353	73	1289	3.0	318	73	1632	2.4	295
	6,000					71	872	6.0	440	71	1207	4.6	382	71	1564	3.6	345
	8,000													69	1466	4.9	397
	10,000																
	12,000																
+55 DEGREES	SL	76	721	0	200	76	998	0	200	76	1294	0	200	76	1615	0	200
	2,000	75	633	3.0	318	75	924	2.1	283	75	1234	1.6	263	75	1567	1.3	250
	4,000					73	835	4.4	374	73	1160	3.3	330	73	1507	2.6	302
	6,000									71	1070	5.0	402	71	1433	3.9	357
	8,000																
	10,000																
	12,000																

REMARKS: 1. CLIMBS BASED ON 3450 BHP AND 2600 RPM AT SEA LEVEL.
2. FUEL ALLOWANCE FOR WARM-UP AND TAKE-OFF, 200 LB (5 MIN AT 3450 BHP).

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958
FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure A—14. Climb Chart For Normal Power — 72-Foot Main Rotor

CLIMB CHART FOR NORMAL POWER

CONFIGURATION: TWO 150 OR 300 GAL AUXILIARY TANKS: GEAR UP

MODEL: HR2S-1 — 72 FOOT MAIN ROTOR — ENGINES: (2) R-2800-54

TEMPERATURE DEGREES CENTIGRADE	PRESSURE ALTITUDE FEET	GROSS WEIGHT LB															
		31,000				28,500				26,000				23,500			
		BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB	BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB	BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB	BEST IAS KNOTS	RATE OF CLIMB FPM	FROM SL TIME MIN	FROM SL FUEL USED LB
—25 DEGREES	SL	76	1189	0	200	76	1453	0	200	76	1740	0	200	76	2055	0	200
	2,000	75	1137	1.7	269	75	1411	1.4	256	75	1708	1.2	246	75	2031	1.0	239
	4,000	73	1072	3.5	341	73	1357	2.8	314	73	1664	2.4	294	73	1997	2.0	279
	6,000	71	992	5.5	419	71	1290	4.4	374	71	1609	3.6	343	71	1953	3.0	319
	8,000	69	684	7.8	511	69	1208	6.0	438	69	1540	4.8	394	69	1896	4.0	361
	10,000	67	269	11.9	675	67	885	7.8	512	67	1441	6.2	446	67	1811	5.1	404
	12,000									65	968	7.8	507	65	1522	6.3	448
- 5 DEGREES	SL	76	1052	0	200	76	1318	0	200	76	1606	0	200	76	1921	0	200
	2,000	75	992	2.0	278	75	1269	1.5	262	75	1567	1.3	250	75	1891	1.1	242
	4,000	73	918	4.0	362	73	1207	3.2	326	73	1517	2.6	302	73	1852	2.1	285
	6,000	71	828	6.3	453	71	1131	4.9	395	71	1454	3.9	356	71	1801	3.2	328
	8,000	69	470	9.4	575	69	1039	6.7	468	69	1376	5.3	412	69	1737	4.3	374
	10,000									67	1197	6.8	472	67	1568	5.5	421
	12,000													65	1274	6.9	470
+15 DEGREES	SL	76	922	0	200	76	1189	0	200	76	1478	0	200	76	1794	0	200
	2,000	75	854	2.3	290	75	1133	1.7	269	75	1434	1.4	255	75	1759	1.1	245
	4,000	73	770	4.7	389	73	1064	3.5	341	73	1377	2.8	312	73	1714	2.3	291
	6,000	71	607	7.5	500	71	978	5.5	420	71	1306	4.3	371	71	1656	3.5	339
	8,000									69	1219	5.9	434	69	1584	4.7	388
	10,000													67	1344	6.0	440
	12,000																
+35 DEGREES	SL	76	795	0	200	76	1066	0	200	76	1356	0	200	76	1673	0	200
	2,000	75	719	2.6	306	75	1002	1.9	277	75	1305	1.5	260	75	1633	1.2	248
	4,000	73	626	5.6	425	73	924	4.0	360	73	1241	3.1	323	73	1582	2.5	298
	6,000					71	830	6.3	451	71	1162	4.7	389	71	1517	3.7	350
	8,000													69	1421	5.[illegible]	404
	10,000																
	12,000																
+55 DEGREES	SL	76	672	0	200	76	946	0	200	76	1239	0	200	76	1557	0	200
	2,000	75	587	3.2	327	75	875	2.2	288	75	1182	1.7	266	75	1512	1.3	252
	4,000					73	789	4.6	384	73	1111	3.4	336	73	1455	2.7	306
	6,000									71	1024	5.3	411	71	1383	4.1	362
	8,000																
	10,000																
	12,000																

REMARKS: 1. CLIMBS BASED ON 3450 BHP AND 2600 RPM AT SEA LEVEL.
2. FUEL ALLOWANCE FOR WARM-UP AND TAKE-OFF, 200 LB (5 MIN AT 3450 BHP).

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

FUEL GRADE: 115 145
FUEL DENSITY: 6.0 LB GAL

Figure A—15. Climb Chart For Normal Power — 72-Foot Main Rotor

LANDING DISTANCE – FEET
POWER OFF

FIRM DRY SOD
CONFIGURATION: ALL
72 FOOT MAIN ROTOR

MODEL: HR2S-1 ENGINES: (2) R-2800-54

GROSS WEIGHT LB	PRESSURE ALTITUDE FEET	BEST IAS APPROACH KNOTS	–25 DEG CENT.		–5 DEG CENT.		+15 DEG CENT.		+35 DEG CENT.		+55 DEG CENT.	
			GROUND ROLL	CLEAR 50'	GROUND ROLL	CLEAR 50'	GROUND ROLL	CLEAR 50'	GROUND ROLL	CLEAR 50'	GROUND ROLL	CLEAR 50'
31,000	SL	88	254	774	280	800	305	825	325	845	345	865
	2,000	88	282	802	306	826	327	846	349	867	370	890
	4,000	88	307	827	330	849	350	869	373	894		
	6,000	88	328	848	352	871	378	898				
	8,000	88	351	870	379	899						
	10,000	88	380	900								
	12,000	88										
28,500	SL	88	234	754	258	778	280	800	300	820	320	839
	2,000	88	259	779	281	801	302	821	322	842	343	862
	4,000	88	284	802	305	824	324	844	345	865	369	887
	6,000	88	303	822	326	846	349	869	372	890		
	8,000	88	325	845	350	870	375	895				
	10,000	88	351	870	377	896						
	12,000	88	380	898								
26,000	SL	88	219	738	239	758	259	778	276	795	295	814
	2,000	88	240	759	260	779	279	797	296	815	315	835
	4,000	88	260	780	280	800	299	818	316	836	339	859
	6,000	88	279	799	300	820	320	840	341	861	365	885
	8,000	88	300	819	321	841	345	865	370	890		
	10,000	88	323	843	346	866						
	12,000	88	349	867								
23,500	SL	88	200	720	218	738	235	755	251	770	266	785
	2,000	88	219	739	235	755	252	771	269	788	285	805
	4,000	88	237	755	254	772	270	790	286	806	306	826
	6,000	88	253	771	271	791	290	810	309	829	332	852
	8,000	88	270	790	290	811	312	833	337	858		
	10,000	88	291	812	314	834						
	12,000	88	315	835								

REMARKS: POWER ON LANDING DISTANCES WILL BE APPROXIMATELY 10-15% LESS THAN THE DISTANCE STATED ABOVE. AT GROSS WEIGHT, TEMPERATURE AND ALTITUDE CONDITIONS WHERE HOVERING OUT OF GROUND EFFECT IS POSSIBLE, VERTICAL DESCENTS CAN BE MADE. FOR THESE CONDITIONS, SEE FIGURE A-20.

BASED ON: ESTIMATED DATA
DATA AS OF: DECEMBER 1, 1958

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure A–16. Landing Distance – 72-Foot Main Rotor

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 31,000 TO 28,500 POUNDS

CONFIGURATION: CLEAN, GEAR UP

72 FOOT MAIN ROTOR

NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE-value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS-ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES

COLUMN I: NAUTICAL MILES SL	2000'	4000'	6000'	FUEL LB	COLUMN II: NAUTICAL MILES SL	2000'	4000'	6000'	COLUMN III: NAUTICAL MILES SL	2000'	4000'	6000'
					SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT (1)							
117	126	150	133	2400	149	158	153		168	166	154	
103	110	131	116	2100	130	138	134		147	145	135	
88	95	113	99	1800	111	118	115		126	124	115	
73	79	94	83	1500	93	99	95		105	104	96	
59	63	75	66	1200	74	79	76		84	83	77	
44	47	56	49	900	55	59	57		63	62	57	
29	31	37	33	600	37	39	38		42	41	38	
14	15	18	16	300	18	19	19		21	20	19	

ENDURANCE - HOURS

COLUMN A SL	2000'	4000'	6000'	8000'	FUEL LB	COLUMN B SL	2000'	4000'	6000'	8000'
1.0	1.0	1.0	1.0		2400	2.0	2.0	1.9	1.7	
0.9	0.9	0.9	0.9		2100	1.8	1.7	1.7	1.5	
0.7	0.7	0.7	0.7		1800	1.5	1.5	1.4	1.3	
0.6	0.6	0.6	0.6		1500	1.2	1.2	1.2	1.1	
0.5	0.5	0.5	0.5		1200	1.0	1.0	0.9	0.8	
0.3	0.3	0.3	0.3		900	0.7	0.7	0.7	0.6	
0.2	0.2	0.2	0.2		600	0.5	0.5	0.4	0.4	
0.1	0.1	0.1	0.1		300	0.2	0.2	0.2	0.2	

Power settings – Range

MAXIMUM CONTINUOUS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	PRESS ALT FEET	MEDIUM CRUISE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	MAXIMUM AIR RANGE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	33.1	N	1370	66	76	6000												
2600	36.6	N	1560	90	98	4000	2600	35.8	N	1500	87	96	2600	35.1	N	1460	85	94
2600	42.5	N	2120	105	112	2000	2600	37.4	N	1560	97	103	2500	35.0	N	1355	88	94
2600	45.8	N	2402	114	118	SL	2600	40.1	N	1771	107	110	2500	37.3	N	1435	98	101

Power settings – Endurance

HOVERING — MINIMUM TAS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	PRESS ALT FEET	MAXIMUM ENDURANCE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2500	44.1	N	2322	20	27.5	6000	2600	32.5	N	1340	58	68
2500	44.9	N	2322	19	25.0	4000	2500	32.4	N	1215	62	70
2500	45.6	N	2322	16	22.0	2000	2500	32.5	N	1175	62	68
2500	46.1	N	2322	15	20.0	SL	2500	32.5	N	1160	62	66

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. A-13) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 31,000 lb gross weight with 1500 lb of fuel (after deducting total allowances of 285 lb) to fly 100 nautical airmiles at 2000 ft altitude maintain 2500 rpm and 35.0 in manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 31,000 lb gross weight with 1500 lb of fuel (after deducting total allowances of 285 lb) to stay in the air 1.2 hours at 2000 ft altitude maintain 2500 rpm and 32.5 in manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle
AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: ___ Knots
N. Normal

FUEL GRADE: 115/145

FUEL DENSITY: 6.0 LB/GAL

DATA AS OF: DECEMBER 1, 1958

BASED ON: FLIGHT TEST

Figure No. A—17. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 1)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 28,500 TO 26,000 POUNDS

CONFIGURATION: CLEAN, GEAR UP
72 FOOT MAIN ROTOR
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES

SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT (1)

COLUMN I — NAUTICAL MILES SL	2000'	4000'	6000'	FUEL LB	COLUMN II — NAUTICAL MILES SL	2000'	4000'	6000'	COLUMN III — NAUTICAL MILES SL	2000'	4000'	6000'
123	121	152	162	2400	163	163	163	166	178	181	179	168
108	106	133	142	2100	142	142	142	145	155	158	157	147
92	91	114	122	1800	122	122	122	125	133	135	134	126
77	76	95	101	1500	102	102	102	104	111	113	112	105
61	60	76	81	1200	81	81	81	83	89	90	89	84
46	45	57	61	900	61	61	61	62	66	67	67	63
31	30	38	40	600	40	40	40	41	44	45	44	42
15	15	19	20	300	20	20	20	20	22	22	22	21

MAXIMUM CONTINUOUS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	PRESS ALT FEET	MEDIUM CRUISE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	MAXIMUM AIR RANGE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	35.1	N	1490	89	101	6000	2600	33.9	N	1410	86	98	2600	33.0	N	1350	84	95
2600	39.8	N	1790	104	114	4000	2600	35.9	N	1505	97	107	2500	34.4	N	1335	91	100
2600	45.0	N	2402	115	122	2000	2500	40.0	N	1662	106	113	2500	36.2	N	1390	99	105
2600	45.8	N	2402	120	124	SL	2500	40.1	N	1676	111	114	2500	36.5	N	1415	103	105

ENDURANCE - HOURS

COLUMN A SL	2000'	4000'	6000'	8000'	FUEL LB	COLUMN B SL	2000'	4000'	6000'	8000'
1.0	1.0	1.0	1.0	1.0	2400	2.2	2.1	2.1	2.1	1.9
0.9	0.9	0.8	0.8	0.8	2100	1.9	1.9	1.8	1.8	1.7
0.7	0.7	0.7	0.7	0.7	1800	1.6	1.6	1.6	1.5	1.4
0.6	0.6	0.6	0.6	0.6	1500	1.3	1.3	1.3	1.3	1.2
0.5	0.5	0.5	0.5	0.5	1200	1.1	1.0	1.0	1.0	0.9
0.3	0.3	0.3	0.3	0.3	900	0.8	0.8	0.8	0.7	0.7
0.2	0.2	0.2	0.2	0.2	600	0.5	0.5	0.5	0.5	0.4
0.1	0.1	0.1	0.1	0.1	300	0.2	0.2	0.2	0.2	0.2

HOVERING — MINIMUM TAS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	PRESS ALT FEET	MAXIMUM ENDURANCE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	43.0	N	2402	14	22.0	8000	2600	30.3	N	1220	57	69
2600	43.6	N	2402	8	14.0	6000	2500	30.0	N	1135	62	72
2600	44.2	N	2402	9	14.0	4000	2500	30.1	N	1115	62	70
2500	45.6	N	2322	8	13.0	2000	2500	30.4	N	1095	62	68
2500	46.1	N	2322	6	10.0	SL	2500	30.5	N	1080	62	66

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. A-13) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 28,500 lb gross weight with 1500 lb of fuel (after deducting total allowances of 265 lb) to fly 75 nautical airmiles at 2000 ft altitude maintain 2600 rpm and 45.0 in manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 28,500 lb gross weight with 1500 lb of fuel (after deducting total allowances of 265 lb) to stay in the air 1.3 hours at 2000 ft altitude maintain 2500 rpm and 30.4 in manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle

AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: ___ Knots
N: Normal

DATA AS OF: DECEMBER 1, 1958

BASED ON: FLIGHT TEST

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure No. A—17. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 2)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 26,000 TO 23,500 POUNDS

CONFIGURATION: CLEAN, GEAR UP
72 FOOT MAIN ROTOR
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES													ENDURANCE - HOURS										
COLUMN I NAUTICAL MILES				FUEL LB	COLUMN II NAUTICAL MILES				COLUMN III NAUTICAL MILES				COLUMN A					FUEL LB	COLUMN B				
					SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT(1)																		
SL	2000'	4000'	6000'		SL	2000'	4000'	6000'	SL	2000'	4000'	6000'	SL	2000'	4000'	6000'	8000'		SL	2000'	4000'	6000'	8000'
131	129	140	164	2400	177	179	179	186	189	194	197	199	1.1	1.1	1.0	1.0	1.0	2400	2.4	2.4	2.3	2.3	2.2
115	113	123	143	2100	155	157	157	162	165	169	172	174	1.0	0.9	0.9	0.9	0.8	2100	2.1	2.1	2.0	2.0	2.0
98	97	105	123	1800	133	134	134	139	142	145	147	149	0.8	0.8	0.7	0.7	0.7	1800	1.8	1.8	1.7	1.7	1.7
82	81	87	102	1500	111	112	112	116	118	121	123	124	0.7	0.6	0.6	0.6	0.6	1500	1.5	1.5	1.4	1.4	1.4
65	64	70	82	1200	88	89	89	93	94	97	98	99	0.5	0.5	0.5	0.5	0.5	1200	1.2	1.2	1.1	1.1	1.1
49	48	52	61	900	66	67	67	69	71	72	73	74	0.4	0.4	0.3	0.3	0.3	900	0.9	0.9	0.8	0.8	0.8
33	32	35	41	600	44	44	44	46	47	48	49	49	0.2	0.2	0.2	0.2	0.2	600	0.6	0.6	0.5	0.5	0.5
16	16	17	20	300	22	22	22	23	23	24	24	24	0.1	0.1	0.1	0.1	0.1	300	0.3	0.3	0.2	0.2	0.2

MAXIMUM CONTINUOUS						PRESS ALT FEET	MEDIUM CRUISE						MAXIMUM AIR RANGE					
RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	37.9	N	1710	104	117	6000	2600	34.0	N	1405	96	109	2500	32.8	N	1225	90	102
2600	42.8	N	2135	114	125	4000	2600	35.8	N	1524	104	114	2500	33.1	N	1265	95	104
2600	45.0	N	2402	122	130	2000	2500	38.5	N	1562	110	117	2500	34.1	N	1285	98	104
2600	45.8	N	2402	127	132	SL	2500	39.0	N	1568	112	116	2500	34.1	N	1280	98	101

HOVERING — MINIMUM TAS						PRESS ALT FEET	MAXIMUM ENDURANCE					
RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	43.0	N	2402	5	9.0	8000	2500	27.5	N	1050	59	71
2500	44.1	N	2322	3	5.5	6000	2500	27.6	N	1020	59	69
2500	44.9	N	2322	0	0	4000	2500	28.0	N	1010	58	66
2500	44.1	N	2180	0	0	2000	2500	28.1	N	1000	57	63
2500	43.7	N	2063	0	0	SL	2500	28.5	N	980	56	60

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. A-13) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 26,000 lb gross weight with 1500 lb of fuel (after deducting total allowances of 253 lb) to fly 112 nautical airmiles at 2000 ft altitude maintain 2500 rpm and 38.5 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 26,000 lb gross weight with 1500 lb of fuel (after deducting total allowances of 253 lb) to stay in the air 1.5 hours at 2000 ft altitude maintain 2500 rpm and 28.1 in manifold pressure with mixture set. NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle

AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: Knots
N: Normal

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

DATA AS OF: DECEMBER 1, 1958 BASED ON: FLIGHT TEST

Figure No. A–17. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 3)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 31,000 TO 28,500 POUNDS

CONFIGURATION:
ONE 150 OR 300 GAL. AUXILIARY TANK
GEAR UP
72 FOOT MAIN ROTOR
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT (1)

RANGE - AIR MILES

Column I Nautical Miles SL	2000'	4000'	6000'	FUEL LB	Column II Nautical Miles SL	2000'	4000'	6000'	Column III Nautical Miles SL	2000'	4000'	6000'
202	201	242	228	4200	243	252	252		278	274	259	
183	182	219	206	3800	220	228	228		251	248	234	
164	162	196	184	3400	197	204	204		225	222	210	
144	143	173	163	3000	174	180	180		198	196	185	
125	124	150	141	2600	151	156	156		172	169	160	
106	105	127	119	2200	127	132	132		145	143	135	
86	86	104	97	1800	104	108	108		119	117	111	
67	67	80	76	1400	81	84	84		92	91	86	
48	47	57	54	1000	58	60	60		66	65	61	
28	28	34	32	600	34	36	36		39	39	37	
9	9	11	10	200	11	12	12		13	13	12	

ENDURANCE - HOURS

Column A SL	2000'	4000'	6000'	8000'	FUEL LB	Column B SL	2000'	4000'	6000'	8000'
1.8	1.8	1.8	1.8		4200	3.5	3.4	3.3	3.1	
1.6	1.6	1.6	1.6		3800	3.1	3.1	3.0	2.8	
1.4	1.4	1.4	1.4		3400	2.8	2.8	2.7	2.5	
1.2	1.2	1.2	1.2		3000	2.5	2.4	2.4	2.2	
1.1	1.1	1.1	1.1		2600	2.1	2.1	2.0	1.9	
0.9	0.9	0.9	0.9		2200	1.8	1.8	1.7	1.6	
0.7	0.7	0.7	0.7		1800	1.5	1.4	1.4	1.3	
0.6	0.6	0.6	0.6		1400	1.1	1.1	1.1	1.0	
0.4	0.4	0.4	0.4		1000	0.8	0.8	0.8	0.7	
0.2	0.2	0.2	0.2		600	0.5	0.4	0.4	0.4	
					200	0.1	0.1	0.1	0.1	

Range power settings

PRESS ALT FEET	Maximum Continuous RPM	MP Inches	Mixture	Approx. TOT LB/HR	IAS Knots	TAS Knots	Medium Cruise RPM	MP Inches	Mixture	Approx. TOT LB/HR	IAS Knots	TAS Knots	Maximum Air Range RPM	MP Inches	Mixture	Approx. TOT LB/HR	IAS Knots	TAS Knots
6000	2600	33.6	N	1397	66	76												
4000	2600	37.1	N	1696	89	98	2600	35.7	N	1561	85	94	2600	34.5	N	1456	81	90
2000	2600	43.5	N	2296	103	110	2600	38.4	N	1700	96	102	2500	36.5	N	1453	89	95
SL	2600	45.8	N	2402	113	116	2500	42.0	N	1842	104	107	2500	37.4	N	1478	95	98

Endurance power settings

PRESS ALT FEET	Hovering — Minimum TAS RPM	MP Inches	Mixture	Approx. TOT LB/HR	IAS Knots	TAS Knots	Maximum Endurance RPM	MP Inches	Mixture	Approx. TOT LB/HR	IAS Knots	TAS Knots
6000	2500	44.1	N	2322	20	27	2600	32.9	N	1320	56	66
4000	2500	44.9	N	2322	17	24	2500	32.5	N	1246	62	70
2000	2500	45.6	N	2322	16	22	2500	32.5	N	1212	62	68
SL	2500	46.1	N	2322	15	20	2500	32.9	N	1189	62	66

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. 14) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 31,000 lb gross weight with 3000 lb of fuel (after deducting total allowances of 287 lb) to fly 196 nautical airmiles at 2000 ft altitude maintain 2500 rpm and 36.5 in manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 31,000 lb gross weight with 3000 lb of fuel (after deducting total allowances of 287 lb) to stay in the air 2.4 hours at 2000 ft altitude maintain 2500 rpm and 32.5 in manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle

AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: ___ Knots
N: Normal

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

DATA AS OF: DECEMBER 1, 1958 BASED ON: FLIGHT TEST

Figure No. A—18. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 1)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 28,500 TO 26,000 POUNDS

CONFIGURATION:
ONE 150 OR 300 GAL. AUXILIARY TANK
GEAR UP
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES													ENDURANCE - HOURS										
COLUMN I				FUEL	COLUMN II				COLUMN III				COLUMN A					FUEL	COLUMN B				
NAUTICAL MILES				LB	NAUTICAL MILES				NAUTICAL MILES									LB					
					SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT(1)																		
SL	2000'	4000'	6000'		SL	2000'	4000'	6000'	SL	2000'	4000'	6000'	SL	2000'	4000'	6000'	8000'		SL	2000'	4000'	6000'	8000'
211	206	239	272	4200	277	279	281	281	299	303	300	286	1.8	1.8	1.8	1.7	1.7	4200	3.8	3.8	3.7	3.6	3.4
191	186	216	246	3800	250	252	254	254	271	274	271	259	1.6	1.6	1.6	1.5	1.5	3800	3.5	3.4	3.3	3.3	3.1
171	167	193	220	3400	224	226	227	228	242	245	243	232	1.4	1.4	1.4	1.4	1.4	3400	3.1	3.1	3.0	2.9	2.8
151	147	171	194	3000	197	199	201	201	213	216	214	204	1.2	1.2	1.2	1.2	1.2	3000	2.7	2.7	2.6	2.6	2.4
130	127	148	168	2600	171	172	174	174	185	188	185	177	1.1	1.1	1.1	1.0	1.0	2600	2.4	2.3	2.3	2.2	2.1
110	108	125	142	2200	145	146	147	147	156	159	157	150	0.9	0.9	0.9	0.9	0.9	2200	2.0	2.0	1.9	1.9	1.8
90	88	102	116	1800	118	119	120	120	128	130	128	122	0.7	0.7	0.7	0.7	0.7	1800	1.6	1.6	1.6	1.5	1.4
70	68	79	90	1400	92	93	93	93	99	101	100	95	0.6	0.6	0.6	0.5	0.5	1400	1.2	1.2	1.2	1.2	1.1
50	49	57	64	1000	65	66	67	67	71	72	71	68	0.4	0.4	0.4	0.4	0.4	1000	0.9	0.9	0.8	0.8	0.8
30	29	34	38	600	39	39	40	40	42	43	42	40	0.2	0.2	0.2	0.2	0.2	600	0.5	0.5	0.5	0.5	0.4
10	9	11	12	200	13	13	13	13	14	14	14	13						200	0.1	0.1	0.1	0.1	0.1

MAXIMUM CONTINUOUS						PRESS ALT FEET	MEDIUM CRUISE						MAXIMUM AIR RANGE					
RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
						8000												
2600	35.6	N	1543	88	100	6000	2600	34.0	N	1431	84	96	2600	33.0	N	1362	82	93
2600	41.0	N	1965	102	112	4000	2600	36.4	N	1552	95	104	2500	33.9	N	1329	86	95
2600	45.0	N	2402	111	118	2000	2500	38.8	N	1624	102	108	2500	35.2	N	1369	93	99
2600	45.8	N	2402	117	121	SL	2500	39.5	N	1652	106	109	2500	36.0	N	1374	95	98

HOVERING — MINIMUM TAS						PRESS ALT FEET	MAXIMUM ENDURANCE					
RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	43.0	N	2402	15	22	8000	2600	30.6	N	1206	56	68
2600	43.6	N	2402	11	18	6000	2500	30.2	N	1148	60	70
2500	44.9	N	2322	9	15	4000	2500	30.3	N	1124	58	66
2500	45.6	N	2322	7	12	2000	2500	30.5	N	1090	58	64
2500	46.1	N	2322	5	9	SL	2500	30.8	N	1079	60	64

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. 14) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 28,500 lb gross weight with 3000 lb of fuel (after deducting total allowances of 267 lb) to fly 147 nautical airmiles at 2000 ft altitude maintain 2600 rpm and 45.0 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 28,500 lb gross weight with 3000 lb of fuel (after deducting total allowances of 267 lb) to stay in the air 2.7 hours at 2000 ft altitude maintain 2500 rpm and 30.5 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle

AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: ___ Knots
N: Normal

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

DATA AS OF: DECEMBER 1, 1958 BASED ON: FLIGHT TEST

Figure No. A—18. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 2)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 26,000 TO 23,500 POUNDS

CONFIGURATION:
ONE 150 OR 300 GAL. AUXILIARY TANK
GEAR UP
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES

SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT(1)

COLUMN I NAUTICAL MILES SL	2000'	4000'	6000'	FUEL LB	COLUMN II NAUTICAL MILES SL	2000'	4000'	6000'	COLUMN III NAUTICAL MILES SL	2000'	4000'	6000'
169	169	169	216	3280	225	231	229	245	246	253	256	259
154	154	154	198	3000	205	211	210	224	225	231	235	237
134	134	134	171	2600	178	183	182	194	195	200	203	205
113	113	113	145	2200	150	155	154	164	165	169	172	173
92	92	92	118	1800	123	126	126	134	135	138	140	142
72	72	72	92	1400	96	98	98	104	105	108	109	110
51	51	51	66	1000	68	70	70	74	75	77	78	79
30	30	30	39	600	41	42	42	44	45	46	46	47
10	10	10	13	200	13	14	14	14	15	15	15	15

ENDURANCE - HOURS

COLUMN A SL	2000'	4000'	6000'	8000'	FUEL LB	COLUMN B SL	2000'	4000'	6000'	8000'
1.5	1.5	1.4	1.4	1.3	3280	3.2	3.2	3.2	3.1	3.1
1.4	1.3	1.2	1.2	1.2	3000	2.9	3.0	2.9	2.9	2.9
1.2	1.1	1.1	1.1	1.0	2600	2.5	2.6	2.5	2.5	2.5
1.0	1.0	0.9	0.9	0.9	2200	2.1	2.2	2.1	2.1	2.1
0.8	0.8	0.7	0.7	0.7	1800	1.7	1.8	1.7	1.7	1.7
0.6	0.6	0.6	0.6	0.5	1400	1.3	1.4	1.3	1.3	1.3
0.4	0.4	0.4	0.4	0.4	1000	0.9	1.0	0.9	0.9	0.9
0.2	0.2	0.2	0.2	0.2	600	0.5	0.6	0.5	0.5	0.5
					200	0.1	0.2	0.1	0.1	0.1

POWER SETTINGS (RANGE)

PRESS ALT FEET	MAXIMUM CONTINUOUS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	MEDIUM CRUISE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	MAXIMUM AIR RANGE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
6000	2600	38.2	N	1727	101	114	2600	33.6	N	1387	92	104	2500	31.5	N	1215	85	96
4000	2600	44.2	N	2402	113	124	2500	37.5	N	1600	102	112	2500	33.1	N	1278	91	100
2000	2600	45.0	N	2402	117	124	2500	37.9	N	1589	105	112	2500	34.0	N	1295	94	100
SL	2600	45.8	N	2402	120	124	2500	38.7	N	1618	108	111	2500	35.0	N	1316	96	99

POWER SETTINGS (ENDURANCE)

HOVERING—MINIMUM TAS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	PRESS ALT FEET	MAXIMUM ENDURANCE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	43.0	N	2402	5	9.5	8000	2500	28.0	N	1027	56	68
2500	44.1	N	2322	2	5	6000	2500	28.0	N	1028	56	66
2500	44.9	N	2322	0	1	4000	2500	28.0	N	1017	56	64
2500	44.1	N	2180	0	0	2000	2500	28.4	N	995	56	62
2500	43.7	N	2062	0	0	SL	2500	28.6	N	1003	56	60

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. 14) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 26,000 lb gross weight with 3000 lb of fuel (after deducting total allowances of 254 lb) to fly 211 nautical airmiles at 2000 ft altitude maintain 2500 rpm and 37.9 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 26,000 lb gross weight with 3000 lb of fuel (after deducting total allowances of 254 lb) to stay in the air 3.0 hours at 2000 ft altitude maintain 2500 rpm and 28.4 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle

AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: ___ Knots
N: Normal

DATA AS OF: DECEMBER 1, 1958

BASED ON: FLIGHT TEST

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure No. A—18. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 3)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 31,000 TO 28,500 POUNDS

CONFIGURATION:
TWO 150 OR 300 GAL.
AUXILIARY TANKS, GEAR UP
72 FOOT MAIN ROTOR
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES

SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT(1) (Columns II and III)

COLUMN I NAUTICAL MILES SL	2000'	4000'	6000'	FUEL LB	COLUMN II NAUTICAL MILES SL	2000'	4000'	6000'	COLUMN III NAUTICAL MILES SL	2000'	4000'	6000'
274	269	331	306	6000	354	346	352		384	375	361	
251	247	303	280	5500	325	317	322		352	344	331	
229	224	276	255	5000	295	288	293		320	312	301	
206	202	248	229	4500	265	259	264		288	281	271	
183	179	220	204	4000	236	230	234		256	250	241	
160	157	193	178	3500	206	201	205		224	219	210	
137	134	165	153	3000	177	173	176		192	187	180	
114	112	138	127	2500	147	144	146		160	156	150	
91	89	110	102	2000	118	115	117		128	125	120	
68	67	82	76	1500	88	86	88		96	93	90	
45	44	55	51	1000	59	57	58		64	62	60	
22	22	27	25	500	29	28	29		32	31	30	

ENDURANCE - HOURS

COLUMN A SL	2000'	4000'	6000'	8000'	FUEL LB	COLUMN B SL	2000'	4000'	6000'	8000'
2.5	2.5	2.5	2.5		6000	4.9	4.9	4.7	4.4	
2.3	2.3	2.3	2.3		5500	4.5	4.5	4.3	4.1	
2.1	2.1	2.1	2.1		5000	4.1	4.1	3.9	3.7	
1.9	1.9	1.9	1.9		4500	3.7	3.7	3.5	3.3	
1.7	1.7	1.7	1.7		4000	3.3	3.2	3.1	2.9	
1.5	1.5	1.5	1.5		3500	2.8	2.8	2.7	2.6	
1.2	1.2	1.2	1.2		3000	2.4	2.4	2.3	2.2	
1.0	1.0	1.0	1.0		2500	2.0	2.0	1.9	1.8	
0.8	0.8	0.8	0.8		2000	1.6	1.6	1.5	1.4	
0.6	0.6	0.6	0.6		1500	1.2	1.2	1.1	1.1	
0.4	0.4	0.4	0.4		1000	0.8	0.8	0.7	0.7	
0.2	0.2	0.2	0.2		500	0.4	0.4	0.3	0.3	

RANGE POWER SETTINGS

MAXIMUM CONTINUOUS			APPROXIMATE			PRESS ALT FEET	MEDIUM CRUISE			APPROXIMATE			MAXIMUM AIR RANGE			APPROXIMATE		
RPM	MP INCHES	MIX-TURE	TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	TOT LB/HR	KNOTS IAS	KNOTS TAS	RPM	MP INCHES	MIX-TURE	TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	33.5	N	1370	60	70	6000												
2600	37.7	N	1703	85	94	4000	2600	35.4	N	1516	82	89	2600	34.0	N	1410	77	85
2600	45.0	N	2402	102	108	2000	2600	39.3	N	1733	94	100	2500	36.5	N	1454	85	91
2600	45.8	N	2402	108	110	SL	2500	40.0	N	1709	98	101	2500	37.2	N	1450	90	93

ENDURANCE POWER SETTINGS

HOVERING — MINIMUM TAS			APPROXIMATE			PRESS ALT FEET	MAXIMUM ENDURANCE			APPROXIMATE		
RPM	MP INCHES	MIX-TURE	TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	TOT LB/HR	KNOTS IAS	KNOTS TAS
2500	44.1	N	2322	20	28	6000	2600	33.4	N	1340	55	65
2500	44.9	N	2322	18	25	4000	2500	32.9	N	1266	61	69
2500	45.6	N	2322	16	22	2000	2500	33.0	N	1216	62	68
2500	46.1	N	2322	14	20	SL	2500	33.1	N	1208	60	64

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. A-15) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 31,000 lb gross weight with 4500 lb of fuel (after deducting total allowances of 289 lb) to fly 281 nautical airmiles at 2000 ft altitude maintain 2500 rpm and 36.5 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 31,000 lb gross weight with 4500 lb of fuel (after deducting total allowances of 289 lb) to stay in the air 3.7 hours at 2000 ft altitude maintain 2500 rpm and 33.0 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle
AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: ___ Knots
N: Normal

DATA AS OF: DECEMBER 1, 1958 BASED ON: FLIGHT TEST

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure No. A–19. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 1)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 28,500 TO 26,000 POUNDS

CONFIGURATION:
TWO 150 OR 300 GAL. AUXILIARY TANKS
GEAR UP
72 FOOT MAIN ROTOR
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES

COLUMN I NAUTICAL MILES SL	2000'	4000'	6000'	FUEL LB	COLUMN II NAUTICAL MILES SL	2000'	4000'	6000'	COLUMN III NAUTICAL MILES SL	2000'	4000'	6000'
					SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT(1)							
266	264	275	331	5518	347	347	347	358	376	382	381	365
241	239	250	300	5000	314	314	314	325	341	346	345	330
217	215	225	270	4500	283	283	283	292	307	311	311	297
193	191	200	240	4000	251	251	251	260	273	277	276	264
169	167	175	210	3500	220	220	220	227	239	242	241	231
144	143	150	180	3000	188	188	188	195	204	207	207	198
120	119	125	150	2500	157	157	157	162	170	173	172	165
96	95	100	120	2000	125	125	125	130	136	138	138	132
72	71	75	90	1500	94	94	94	97	102	103	103	99
48	47	50	60	1000	62	62	62	65	68	69	69	66
24	23	25	30	500	31	31	31	32	34	34	34	33

MAXIMUM CONTINUOUS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	PRESS ALT FEET	MEDIUM CRUISE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	MAXIMUM AIR RANGE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	36.9	N	1650	87	99	6000	2600	34.6	N	1446	83	94	2600	33.2	N	1360	79	90
2600	42.5	N	2200	100	110	4000	2600	36.8	N	1606	92	101	2500	34.2	N	1331	83	92
2600	45.0	N	2402	108	115	2000	2500	38.9	N	1669	99	105	2500	35.4	N	1371	89	95
2600	45.8	N	2402	112	116	SL	2500	39.8	N	1667	102	105	2500	36.0	N	1391	92	95

ENDURANCE - HOURS

COLUMN A SL	2000'	4000'	6000'	8000'	FUEL LB	COLUMN B SL	2000'	4000'	6000'	8000'
2.3	2.3	2.3	2.3	2.2	5518	5.0	4.9	4.8	4.7	4.5
2.1	2.1	2.1	2.1	2.0	5000	4.5	4.5	4.4	4.3	4.1
1.9	1.9	1.9	1.9	1.8	4500	4.1	4.0	3.9	3.9	3.6
1.7	1.7	1.7	1.7	1.6	4000	3.6	3.6	3.5	3.4	3.2
1.5	1.5	1.5	1.5	1.4	3500	3.1	3.1	3.1	3.0	2.8
1.2	1.2	1.2	1.2	1.2	3000	2.7	2.7	2.6	2.6	2.4
1.0	1.0	1.0	1.0	1.0	2500	2.2	2.2	2.2	2.1	2.0
0.8	0.8	0.8	0.8	0.8	2000	1.8	1.8	1.7	1.7	1.6
0.6	0.6	0.6	0.6	0.6	1500	1.3	1.3	1.3	1.3	1.2
0.4	0.4	0.4	0.4	0.4	1000	0.9	0.9	0.8	0.8	0.8
0.2	0.2	0.2	0.2	0.2	500	0.4	0.4	0.4	0.4	0.4

HOVERING — MINIMUM TAS RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	PRESS ALT FEET	MAXIMUM ENDURANCE RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	43.0	N	2402	14	22	8000	2600	31.0	N	1218	55	67
2500	44.1	N	2322	12	18	6000	2500	30.5	N	1151	58	68
2500	44.9	N	2322	9	14	4000	2500	30.9	N	1128	58	66
2500	45.6	N	2322	7	11	2000	2500	30.9	N	1107	57	63
2500	46.1	N	2322	5	9	SL	2500	31.1	N	1095	56	60

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. A-15) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 28,500 lb gross weight with 4500 lb of fuel (after deducting total allowances of 268 lb) to fly 215 nautical airmiles at 2000 ft altitude maintain 2600 rpm and 45.0 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 28,500 lb gross wt with 4500 lb of fuel (after deducting total allowances of 268 lb) to stay in the air 4.0 hours at 2000 ft altitude maintain 2500 rpm and 30.9 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle

AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: —— Knots
N: Normal

DATA AS OF: DECEMBER 1, 1958

BASED ON: FLIGHT TEST

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

Figure No. A—19. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 2)

AIRCRAFT MODEL: HR2S-1

ENGINE: R-2800-54

FLIGHT OPERATION INSTRUCTION CHART

STANDARD DAY

CHART WEIGHT LIMITS: 26,000 TO 23,500 POUNDS

CONFIGURATION:
TWO 150 OR 300 GAL. AUXILIARY TANKS
GEAR UP
72 FOOT MAIN ROTOR
NUMBER OF ENGINES OPERATING: 2

INSTRUCTIONS - RANGE: Select figure in FUEL column equal to or less than amount of fuel to be used for cruising (1). Move horizontally to right or left and select RANGE value equal to or greater than the statute or nautical airmiles to be flown. Vertically below and opposite value nearest desired cruising altitude (ALT) read RPM Manifold pressure, (MP) and MIXTURE setting required. Fuel flow and true air speeds (TAS) are approximate Columns I, II, & III give progressive increase in range at a sacrifice in speed. Range values are for an average airplane flying alone (NO WIND). Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

INSTRUCTIONS - ENDURANCE: Select figure in FUEL column equal to or less than amount of fuel to be used for endurance. Move horizontally to right or left (column A for hovering—column B for max. endurance) and under the desired pressure altitude read hours endurance available. Vertically below and opposite value nearest desired altitude (ALT) read RPM manifold pressure (MP), mixture setting and approximate TAS. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

RANGE - AIR MILES														ENDURANCE - HOURS										
COLUMN I (NAUTICAL MILES)				FUEL LB	COLUMN II (NAUTICAL MILES)				COLUMN III (NAUTICAL MILES)					COLUMN A					FUEL LB	COLUMN B				
					SUBTRACT FUEL ALLOWANCE NOT AVAILABLE FOR FLIGHT (1)																			
SL	2000'	4000'	6000'		SL	2000'	4000'	6000'	SL	2000'	4000'	6000'		SL	2000'	4000'	6000'	8000'		SL	2000'	4000'	6000'	8000'
150	150	150	177	3018	198	198	205	215	219	221	226	227		1.4	1.3	1.2	1.2	1.2	3018	2.9	2.9	2.9	2.9	2.8
124	124	124	147	2500	164	164	170	178	181	183	187	188		1.2	1.1	1.0	1.0	1.0	2500	2.4	2.4	2.4	2.4	2.3
99	99	99	117	2000	131	131	136	142	145	146	150	150		0.9	0.9	0.8	0.8	0.8	2000	1.9	1.9	1.9	1.9	1.8
74	74	74	88	1500	98	98	102	106	109	110	112	112		0.7	0.6	0.6	0.6	0.6	1500	1.4	1.4	1.4	1.4	1.4
49	49	49	58	1000	65	65	68	71	72	73	75	75		0.4	0.4	0.4	0.4	0.4	1000	0.9	0.9	0.9	0.9	0.9
24	24	24	29	500	32	32	34	35	36	36	37	37		0.2	0.2	0.2	0.2	0.2	500	0.4	0.4	0.4	0.4	0.4

MAXIMUM CONTINUOUS						PRESS ALT FEET	MEDIUM CRUISE						MAXIMUM AIR RANGE					
RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS	RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	39.8	N	1900	99	112	6000	2600	34.5	N	1445	91	103	2500	32.5	N	1262	84	95
2600	44.2	N	2402	109	120	4000	2500	37.5	N	1588	99	108	2500	33.1	N	1278	87	96
2600	45.0	N	2402	113	120	2000	2500	39.4	N	1669	103	110	2500	35.1	N	1362	94	100
2600	45.8	N	2402	116	120	SL	2500	39.3	N	1639	105	108	2500	35.1	N	1319	93	96

HOVERING — MINIMUM TAS						PRESS ALT FEET	MAXIMUM ENDURANCE					
RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS		RPM	MP INCHES	MIX-TURE	APPROXIMATE TOT LB/HR	KNOTS IAS	KNOTS TAS
2600	43.0	N	2402	5	9	8000	2500	28.1	N	1061	56	68
2500	44.1	N	2322	2	4	6000	2500	28.5	N	1036	54	64
2500	44.9	N	2322	0	0	4000	2500	28.5	N	1030	54	62
2500	44.1	N	2180	0	0	2000	2500	28.5	N	1022	55	61
2500	43.7	N	2062	0	0	SL	2500	29.0	N	1017	55	59

SPECIAL NOTES

1. Make allowance for warm-up, take-off and climb (see fig. A-15) plus allowance for wind, reserve and combat as required.

EXAMPLE-RANGE

At 26,000 lb gross weight with 2000 lb of fuel (after deducting total allowances of 254 lb) to fly 131 nautical airmiles at 2000 ft altitude maintain 2500 rpm and 39.4 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

EXAMPLE-ENDURANCE

At 26,000 lb gross weight with 2000 lb of fuel (after deducting total allowances of 254 lb) to stay in the air 1.9 hours at 2000 ft altitude maintain 2500 rpm and 28.5 in. manifold pressure with mixture set: NORMAL. Refer to corresponding column and altitude for new power settings when gross weight falls below limits of this chart.

LEGEND

MP: Manifold Pressure
ALT: Pressure Altitude
ML: Manual Lean
CL: Cruising Lean
TAS: True Airspeed
FT: Full Throttle

AR: Auto Rich
FR: Full Rich
SL: Sea Level
LB/HR: Fuel Flow
AL: Auto Lean
KN: ___ Knots
N: Normal

FUEL GRADE: 115/145
FUEL DENSITY: 6.0 LB/GAL

DATA AS OF: DECEMBER 1, 1958

BASED ON: FLIGHT TEST

Figure No. A—19. Flight Operation Instruction Chart — 72-Foot Main Rotor (Sheet 3)

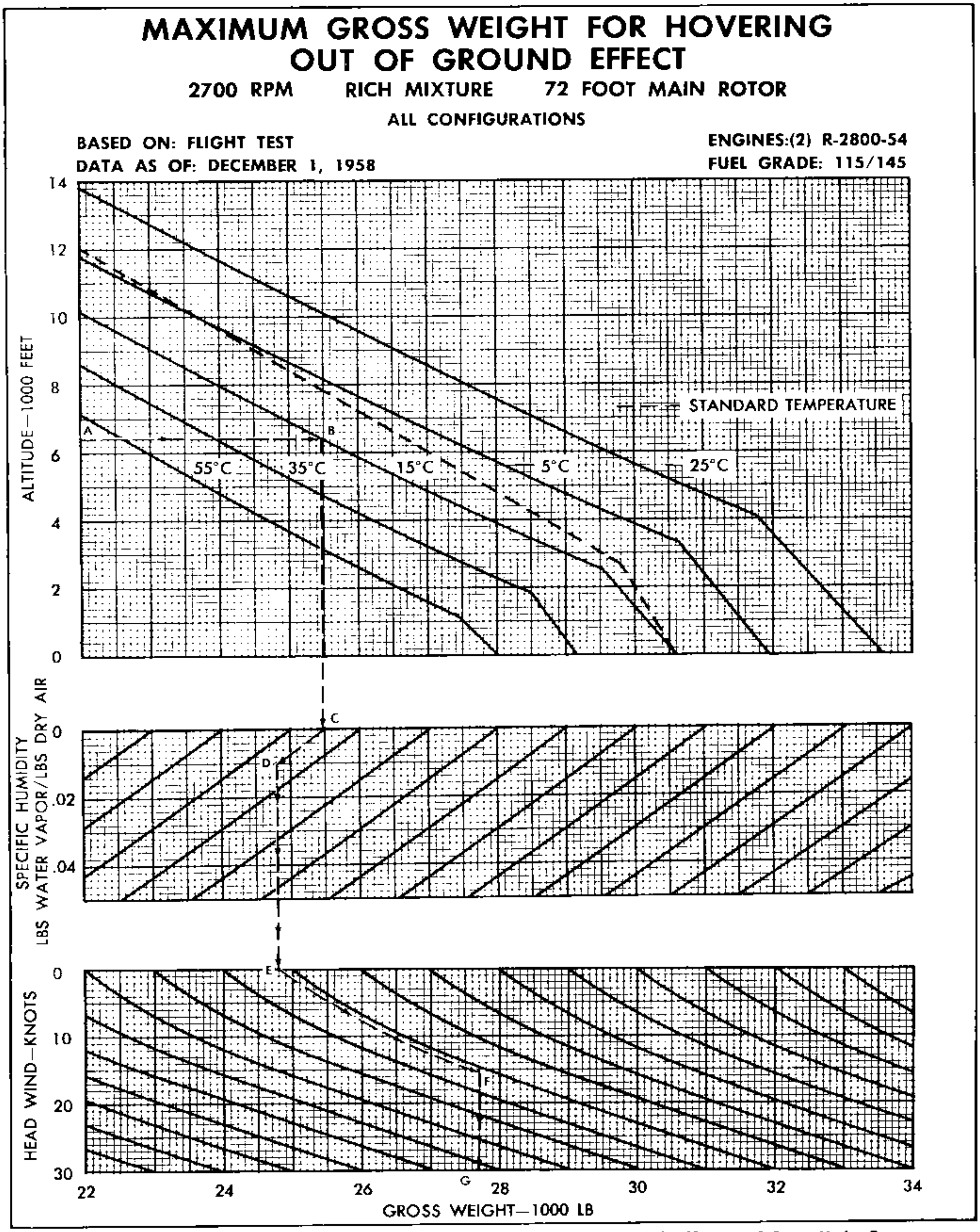


Figure A–20. Maximum Gross Weight for Hovering Out of Ground Effect — 72-Foot Main Rotor

MAXIMUM GROSS WEIGHT FOR HOVERING IN GROUND EFFECT - 10 FOOT WHEEL CLEARANCE

2700 RPM RICH MIXTURE 72 FOOT MAIN ROTOR

ALL CONFIGURATIONS

BASED ON: FLIGHT TEST
DATA AS OF: DECEMBER 1, 1958

ENGINES:(2) R-2800-54
FUEL GRADE: 115/145

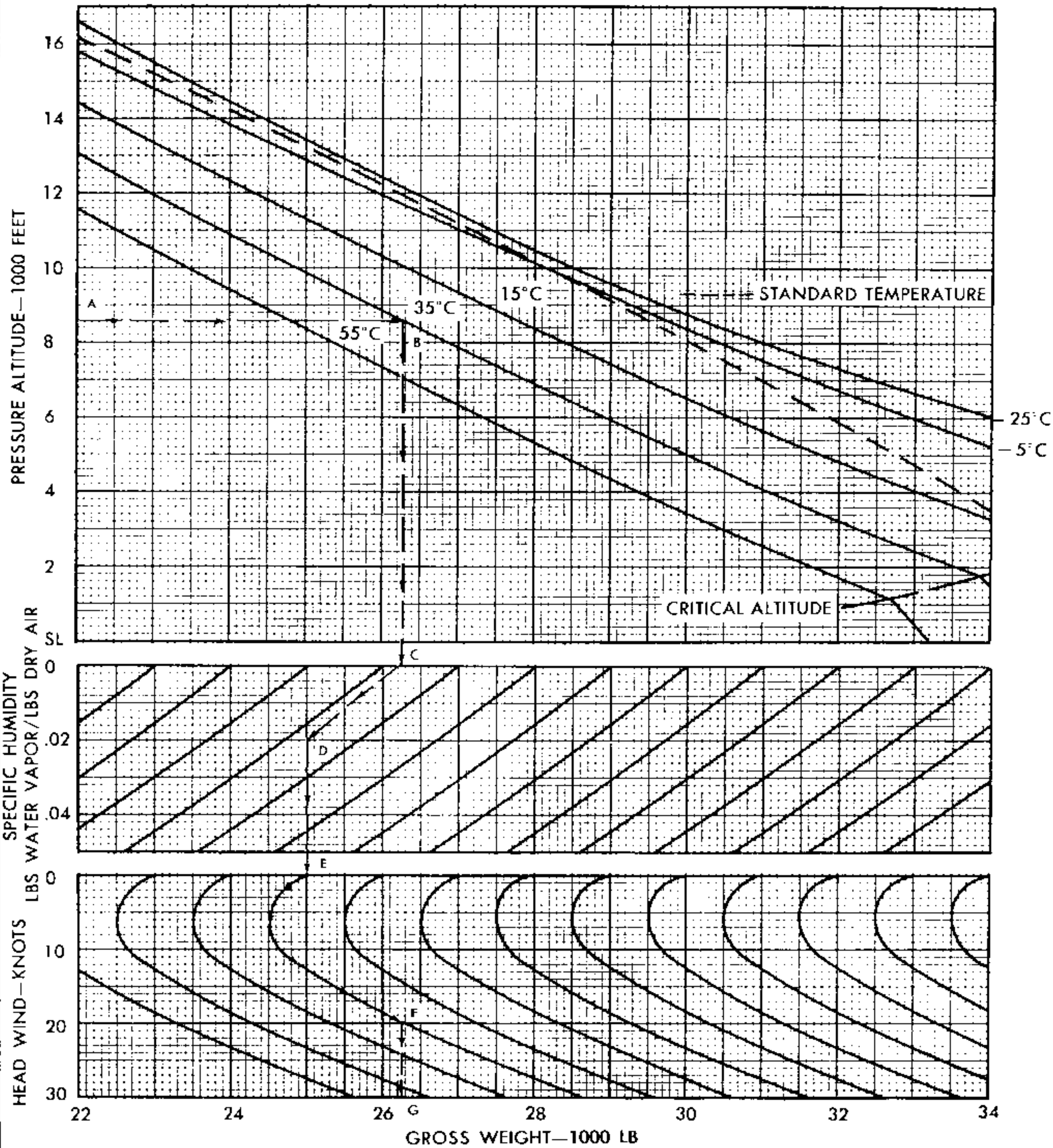


Figure A—21. Maximum Gross Weight for Hovering In Ground Effect — 72-Foot Main Rotor

SPECIFIC HUMIDITY CHART

EXAMPLE

FOR A DRY TEMPERATURE OF 63°F AND A WET BULB TEMPERATURE OF 60°F THE SPECIFIC HUMIDITY IS 0.01

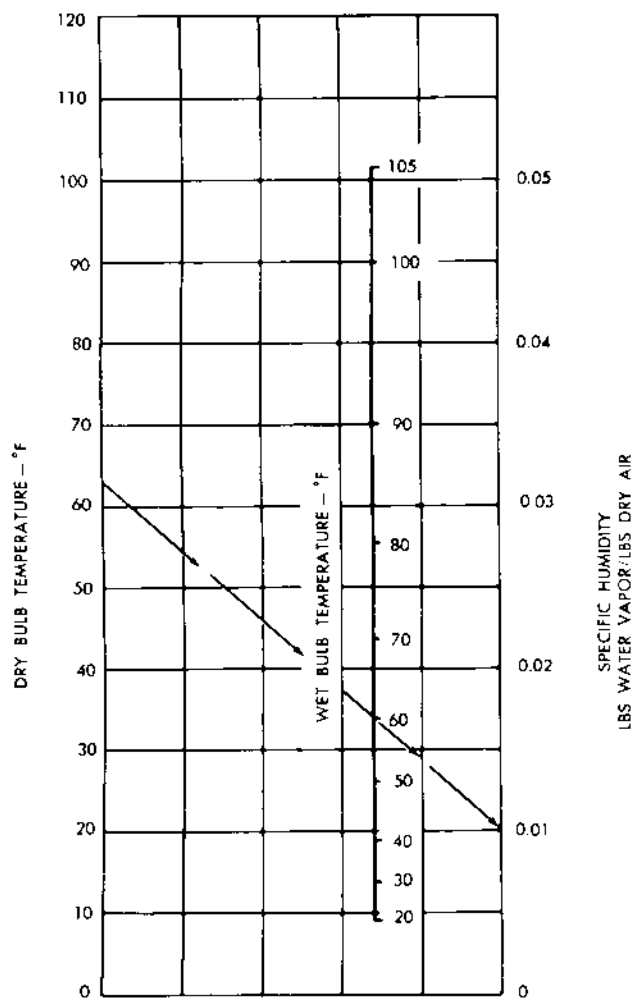


Figure A—22. Specific Humidity Chart

ALPHABETICAL INDEX

(Page numbers in bold face denote illustrations)

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(Page numbers in bold face denote illustrations)