T.O. 1H-3(H)F-1

FLIGHT MANUAL

U.S. COAST GUARD

MODEL HH-3F HELICOPTERS

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This publication replaces T.O. 1H-3(H)F-1 dated 1 August 1974, changed 15 July 1977. Refer to Safety Supplement Index, T.O. 0-1-1-5, for the current status of Flight Manuals, and Safety/Operational Supplements. The Scroll Checklist has not been changed with this revision. The current Scroll Checklist is dated 15 July 1977.

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LIST OF EFFECTIVE PAGES

Insert latest changed pages; dispose of superseded pages in accordance with applicable regulations.

NOTE: On a changed page, the portion of the text affected by the latest change is indicated by a vertical line, or other change symbol, in the outer margin of the page. Changes to illustrations are indicated by miniature pointing hands. Changes to wiring diagrams are indicated by shaded areas.

Total number of pages in this manual is 394 consisting of the following:

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INTRODUCTION

SCOPE

This manual contains the necessary information for safe and efficient operation of the HH-3F helicopter. These instructions provide you with a general knowledge of the helicopter, its characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and therefore, basic flight principles are avoided.

PERMISSIBLE OPERATIONS

The Flight Manual takes a positive approach and normally states only what you can do. Unusual operations or configurations (such as asymmetrical loading) are prohibited unless specifically covered herein.

INTERIM CHANGE SYSTEM

Essential and urgent changes to this technical manual shall be issued by an interim change system. These changes shall be distributed to you on a non-scheduled, as needed basis to disseminate important information at the earliest possible date and eliminate many of your page change collation problems associated with large revisions.

Interim changes shall consist of three types: (1) Brief write-in instructions, (2) supplemental pages, paragraphs, and illustrations, and (3) replacement or additional pages. Write-in instructions indicated must be made directly on the affected page. Supplemental pages must be inserted next to the affected page of the manual as instructed on the change. Replacement and additional pages must be collated into the manual and superseded pages removed. The changes shall be issued in message or printed form. All shall have a change number for control purposes which you must enter on the interim change summary (flyleaf) in this manual.

Periodically all interim changes will be considered to become a permanent part of the basic manual and shall be incorporated as a formal, printed amendment or revision.

WARNINGS, CAUTIONS AND NOTES

The following definitions apply to Warnings, Cautions, and Notes found throughout the manual.

WARNING	Operating procedures, techniques, etc., which will result in personal injury or loss of life if not carefully followed.
CAUTION	Operating procedures, techniques, etc., which will result in damage to equipment if not carefully followed.
NOTE	An operating procedure, technique, etc., which is considered essential to emphasize.

YOUR RESPONSIBILITY - TO LET US KNOW

Every effort is made to keep the Flight Manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusions of the latest data in the manual. However, we cannot correct an error unless we know of its existence. In this regard, it is essential that you do your part. Comments, corrections, and questions regarding this manual or any phase of the Flight Manual program are welcomed. These should be forwarded to the Commanding Officer, U.S. Coast Guard Aviation Training Center (H-3 Training Branch) Mobile, Alabama 36608. Use of A.F. Form 847 is recommended.

Distribution requirements for AF Form 847 are contained in paragraph 4.2.2.2 of the Aeronautical Engineering Maintenance Management Manual (CG-452).

ACTION

Operating commands and individual flight crews shall comply with the procedures and limitations specified in this publication and such duly promulgated changes thereto as may be directed. A copy of this manual shall be carried in each helicopter.

GLOSSARY OF TERMS AND ABBREVIATIONS

AC - Alternating current

ACCELERATION - The rate of change of velocity

ADF - Automatic direction finder

AFCS - Automatic flight control system

ALT - Altitude

APU - Auxiliary power unit

BAR ALT - Barometric altitude control

BLADE TIP STALL - Beginning of blade stall. Occurs at tip of retreating blade due to its high angle of attack and low forward velocity.

BLADE STALL - A stall that begins at the tip of the blade and works progressively inboard as the conditions which cause it increase in severity

FULL BLADE STALL - Blade stall that is allowed to fully develop causing loss of control and an upward left pitch of the helicopter.

INCIPIENT BLADE STALL - Blade tip stall

BOTTOMING - The engine is considered as bottoming during deceleration whenever a minimum fuel flow to compression-discharge pressure condition is attained.

BUOYANCY - The upward force exerted by water on a floating or immersed body by a fluid.

°C - Degrees Celsius

CAS - Calibrated airspeed

CENTER OF GRAVITY (CG) - The center of gravity is the point about which a helicopter would balance if suspended.

COLLECTIVE - The increasing or decreasing of pitch on all the main rotor blades simultaneously. Also short for collective lever.

CYCLIC - The changing of pitch or each main rotor blade individually as it makes a complete rotation or cycle. Also short for cyclic stick.

DC - Direct current

DG - Directional gyro

DRAFT - The depth of water the helicopter draws or requires to float.

DRAG DIVERGENCY - Beginning of blade tip stall.

DROOP - Characteristic built into speed control for speed stability and load sharing. When in the governing range steady state N_f will decrease in proportion to engine load at a fixed N_f setting. On this installation the droop is 8.5% N_f from no load to full load conditions.

DECAY - Loss of N_r beyond droop, resulting from a power requirement in excess of power available.

EXCESS BUOYANCY - Buoyancy in excess of that required to float.

°F - Degrees Fahrenheit

FAT - Free air, ambient, or outside air temperatures

FOD - Foreign object damage

FPM - Feet per minute

FT - Feet

GAL - Gallons

GCA - Ground-controlled approach

GW - Gross weight

HR - Hour

HYDROSTATIC ROLL ANGLE - Angle of roll when helicopter is on the water.

Hz - Hertz (cycles per second)

H/V - Height velocity

IAS - Indicated airspeed

IBIS - In-Flight Blade Inspection System

IGE - In ground effect

IN - Inches

KCAS - Knots calibrated airspeed

KIAS - Knots indicated airspeed

KT - Knots

KTAS - Knots true airspeed

KVA - Kilovolt-amperes

LAT - Latitude

LB/HR - Pound per hour

LOAD FACTOR - A factor representing the ratio of weight or pressure of a specified load or force to a standard weight of pressure. The load factor may represent the ratio of the total weight of the helicopter to a weight or pressure imposed by aerodynamic forces, inertia forces, or ground effect.

MAG - Magnetic

MEAN WATERLINE - The mean of the highest and lowest waterline for a given set of conditions, gross weight, sea state, etc.

MIN - Minutes

MSL - Mean sea level

N_f - Power turbine speed

N_g - Gas generator speed

N_r - Rotor speed

OGE - Out of ground effect. This means hovering at some altitude above the ground or water at which no additional lift is obtained from ground effect.

P_2 - Compressor inlet total pressure

P_3 - Compressor discharge pressure

PRESS - Pressure

PSI - Pounds per square inch

Q - Torque

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RIGHTING MOMENT - A moment that tends to restore the helicopter to a previous position after an angular displacement on the water about one of its axes.

ROC - Rate of climb

ROD - Rate of decent

RPM - Revolutions per minute

SEA STATE - Condition of water surface in terms of wind, wave height, wave length, etc.

SERVICE CEILING - Maximum altitude at which a rate of climb 100 FPM can be maintained.

SL - Sea level

STD DAY - Standard day atmospheric conditions

T_2 - Compressor inlet air temperature FAT may be used in place of T_2 in this manual as T_2 is not indicated in the cockpit.

T_5 - Power turbine inlet temperature

TAS - True airspeed

TEMP - Temperature

TOLD - Take-off and landing data.

TOPPING - A procedure for adjusting engine fuel control to achieve engine performance at maximum operating limits.

TORQUE POWER INDICATION - An indication of power input being delivered to the gear box by the engines.

TRIM ANGLE - The angle at which the helicopter's hull rests on the water.

UTI - Utility

VA - Volt amperes

VAC - Volts alternating current

WATERLINE - The line of intersection between the surface of the water and the side of the helicopter hull when the helicopter is afloat.

WAVE LENGTH - The distance between two successive wave crests.

W_f - Fuel flow

W_f/P_3 - Ratio of weight of fuel flow to be burned to compressor discharge pressure or amount of air available for combustion and cooling.

WL - Water line

SECTION 1

DESCRIPTION

The function of this section is to describe the helicopter and its systems and controls which contribute to the physical act of flying the helicopter, including all emergency equipment that is not part of auxiliary equipment.

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THE HELICOPTER

The helicopter, manufactured by Sikorsky Aircraft, Division of United Technologies Corporation, Stratford, Connecticut, is equipped with a single main rotor, twin engines rated at 1500 SHP each, a fully retractable tricycle landing gear, amphibious capabilities, and hydraulically operated aft ramp that may be opened in flight, on the ground, or on water. The helicopter may be used as a general purpose vehicle to locate, recover, and render assistance to persons in distress. In addition, it may be used for logistic support, reconnaissance, and general utility. The maximum span with the main rotor blades rotating is 62 feet. The maximum length of the helicopter with the rotor blades extended is 73 feet. The height of the highest point of the helicopter (tail rotor) is 18 feet 1 inch. Familiarity with the configuration of the helicopter may be obtained by referring to the exterior and interior general arrangement illustrations, (figures FO-1 and FO-2), and the minimum turning radius and ground clearances diagram (figure 2-4). The helicopter's maximum gross weight is 22,050 pounds. For complete weight information see section V. The fuselage is composed of the cockpit, the upper fuselage, the aft fuselage, the pylon, and the lower fuselage. The upper fuselage section contains the cargo compartment, the engine, transmission and APU compartments. The aft fuselage extends from the cabin to the pylon. The lower fuselage contains four fuel tanks and an electronics rack in the forward section. Sponsons are mounted on each side of the lower fuselage. The pylon is attached to the rear of the aft fuselage. A horizontal stabilizer is mounted on the upper right side of the pylon. The intermediate gear box is installed in the lower portion of the pylon with a shaft extending upward to the tail rotor gear box at the top of the pylon. The five-bladed tail rotor is splined to the tail rotor gear box. The cockpit provides side-by-side seating for the pilot and copilot with the pilot on the right side. To the rear of the cockpit is the cabin. Access between the cockpit and the cabin may be used in flight. A folding jump seat is provided in the cockpit entry. A sliding cargo door is located on the right side of the forward end of the cabin. An eight-foot ramp is located at the rear of the cabin. The cabin accommodates two crewmen and six passengers. Two large windows, located in the forward cabin, will be used as search stations. Swivel type crewman's seats are located adjacent to the search stations. Two electronics racks are located in the cabin; one immediately aft of the copilot and one in the aft portion of the cabin. A folding type navigator's table is mounted on the electronics rack aft of the copilot and forward of the left crewman's seat. Structural provisions are made for additional passenger seats. The cabin is 6.6 feet wide, 6 feet high, and 26 feet 2.5 inches long. Eight feet of the length is ramp area. The cabin is equipped with tiedown rings for transportation of cargo. A 600-pound capacity hydraulic rescue hoist with approximately 240 feet of usable cable is suspended on a fixed truss over the cargo door. Two gas turbine engines are mounted side by side in the engine compartment, which is located above the forward portion of the cabin. The engine drive shafts extend aft into the main gear box which is located in the transmission compartment. The main rotor assembly, to which the five rotor blades are attached, is splined to the main gear box drive shaft. The APU, located aft of the main gear box, is capable of driving the main gear box accessory section and is used for engine starting and checkout of systems. A removable deflector is installed to reduce the possibility of foreign object damage to the engines.

DIMENSIONS

Length.

Maximum main and tail rotor blades extended	73 feet 0 inches
Minimum main and tail rotor blades removed	57 feet 3.53 inches

Height.

Maximum to top of tail rotor blade - vertical static	18 feet 1 inch
Kneeled	20 feet 2 inches
Tail rotor diameter	10 feet 4 inches
Minimum tail rotor blades removed	16 feet 1 inch

Width.

Minimum main rotor blades removed	17 feet 4 inches
Main rotor diameter	62 feet 0 inches
Minimum Main Rotor Ground Clearance. (Tip clearance - forward section)	
Static	10 feet 1 inch
Kneeled	7 feet 4 inches
Tail Rotor Ground Clearance.	
Static	7 feet 9 inches
Kneeled	9 feet 11 inches
Tail Pylon Ground Clearance.	
Static	6 feet 5 inches
Kneeled	8 feet 0 inches
Main Landing Gear Tread.	13 feet 4 inches

ENGINES

The helicopter is equipped with two General Electric T58-GE-5 engines (figures 1-1 and 1-2), each rated at 1500 SHP. The engine is a compact turboshaft engine with high power-to-weight ratio and uses the free turbine principle. The power turbine is mechanically independent of the gas generator and, within the power turbine governing range, power turbine speed is independent of output power. High torque is available at low output speeds, providing rapid acceleration characteristics. The engines are mounted side-by-side above the cargo compartment forward of the main gear box. Each engine consists of the following major components: an axial-flow compressor, combustion chamber, a two-stage gas generator turbine, and a single-stage free power turbine, which is independent of the gas generator turbine. The gas generator consists of the compressor, annular combustor, and two-stage gas generator turbine. The free turbine principle provides a constant free turbine speed output which results in a constant rotor rpm. Variations in power requirements to maintain constant free turbine speed are accomplished by automatic increases or decreases in gas generator speed. A hydromechanical fuel metering unit provides maximum engine performance without

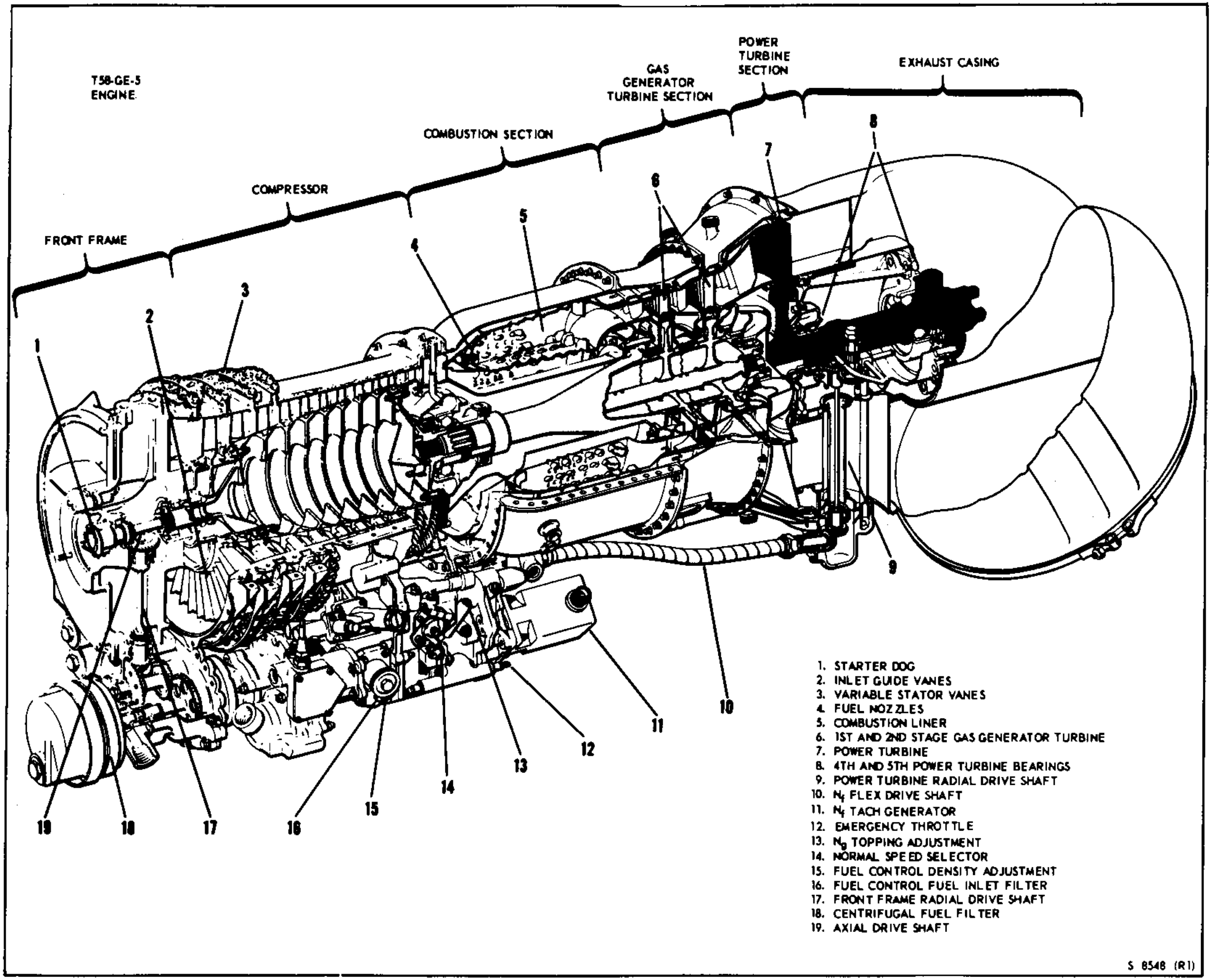

Figure 1-1. Engine Cut-Away View

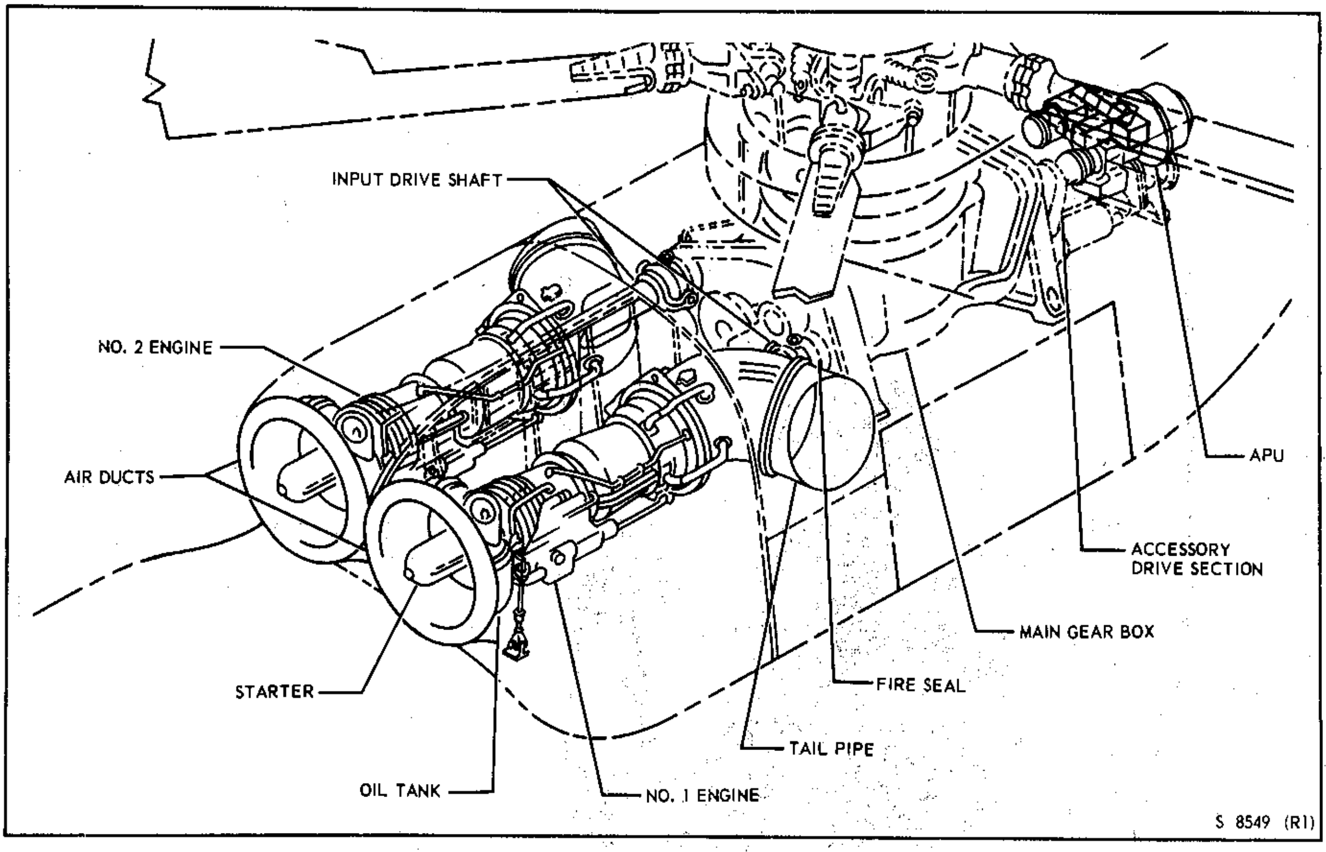

Figure 1-2. Engine, Main Gear Box, and APU Installation

exceeding safe engine operating limits. In the normal operating range, engine speed is selected by positioning the speed selector. The integrated fuel control system delivers atomized fuel in controlled amounts to the combustion chamber. Flow of fuel and air through the combustion chamber is continuous, and once the mixture is ignited, combustion is self-sustained. Changes in air pressure, air temperature, and rotor operation all affect engine performance. With the FOD deflector installed, helicopter velocity has practically no effect on engine performance. The engine fuel control system automatically maintains selected power turbine speed by changing fuel flow to increase or decrease gas generator speed as required, thus regulating output power to match the load under changing conditions. A start bleed valve, mounted on the compressor, automatically opens during the starting cycle to bleed approximately 6.7% of compressor discharge airflow overboard. This decreases compressor discharge pressure, which lessens the possibility of compressor stall and allows the starter to accelerate the gas generator faster. The valve automatically opens when the starter is engaged and remains open until starter dropout. A start fuel valve is installed between the engine fuel control and flow divider. This valve, when closed, shuts off auxiliary starting fuel flow. This valve may be used by the pilot during the starting cycle to reduce starting fuel flow which decreases the possibility of a hot start.

COMPRESSOR

The ten-stage compressor consists of the compressor rotor and stator. The compressor rotor is supported by the front frame section and the compressor rear frame section. The stator is bolted between the front frame section and compressor rear frame. The primary purpose of the compressor is to compress air for combustion. Ambient air enters through the front frame and is directed to the compressor inlet, passes through ten stages of compression, and is directed to the combustion chambers. The inlet guide vanes (2, figure 1-1) and the first three stages of the stator vanes (3, figure 1-1) are variable, and change their angular position as a function of compressor inlet temperature and gas generator speed to prevent stall of the compressor.

COMBUSTION CHAMBER

In the combustion chamber fuel is added to the compressed air. This mixture is ignited, causing a rapid expansion of gases toward the gas generator turbine section. As the air enters the combustion section, a portion goes into the combustion chamber where it is mixed with the fuel and ignited, while the remaining air forms a blanket between the outer combustion casing and the combustion liner (5, figure 1-1), for cooling purposes. Once combustion is started by the two igniter plugs, it is self-sustaining. After the air has been expanded and increased in velocity by combustion, it is passed through the first-stae turbine wheel of the gas generator turbine (6, figure 1-1).

GAS GENERATOR TURBINE

The two-stage gas generator turbine (6, figure 1-1) is the rotating component which is coupled directly to the compressor. It extracts the required power from the exhaust gases to drive the compressor. The turbine nozzles that comprise the stator blades direct the exhaust gases to the turbine wheels.

POWER TURBINE

The power turbine (7, figure 1-1) is bolted to the rear flange of the second stage turbine casing. The engine utilizes the free turbine principle wherein engine output power is provided by the power turbine rotor, which is mechanically independent of the gas generator rotor. This rotor derives its power from the gases which are directed to it by the gas generator turbine nozzles. Within the normal operating range, power turbine speed may be maintained or regulated independent of output power. This principle also provides more rapid acceleration because of the availability of high engine torque at low output speeds.

GAS GENERATOR SPEED (N_g)

Gas generator speed (N_g) is primarily dependent upon fuel flow and is monitored by the engine fuel control unit. The principal purpose of monitoring N_g is to control acceleration and deceleration characteristics, prevent overspeed, and establish a minimum idle setting. Gas generator speed controls mass airflow pumped through the engine and consequently the power available to the power turbine.

FREE POWER TURBINE SPEED (N_f)

The free power turbine speed (N_f) is dependent upon speed selector position and rotor load. The fuel control monitors N_f to regulate fuel flow to maintain an essentially constant power turbine speed for a given speed selector position.

ENGINE FUEL SYSTEM

The engine fuel systems (figure 1-3), one for each engine, consist of an engine-driven pump, a dynamic filter, a fuel control unit, a static filter, an oil cooler, a flow divider, a fuel manifold, and associated piping. The fuel control unit is supplied fuel from the engine-driven fuel pump. Metered fuel from the engine fuel control unit is piped through an oil-fuel heat exchanger and then enters the flow divider connected directly to the fuel manifold on the engine. For normal flight, rotor speed is selected by positioning the speed selectors, and the engine fuel controls will meter fuel to maintain the selected rotor speed.

Engine-Driven Fuel Pump

A dual-operation engine-driven fuel pump, mounted on each engine, consisting of a positive displacement type gear pump and a centrifugal boost pump, is built into a single housing. Power for each pump is furnished from the engine accessory drive section by means of a splined shaft. This shaft drives the fuel pump and simultaneously acts as a link to transmit gas generator speed information to the engine fuel control unit. If the engine-driven fuel pump or splined shaft fails, the engine will flame out due to fuel starvation.

Engine Fuel Control Unit

The engine fuel control units, one located on each engine, are hydromechanical units that regulate engine fuel flow to maintain a selected constant free power turbine speed (N_f) and a constant rotor speed (N_r). Fuel from the engine fuel pump enters the fuel control unit through the inlet and passes through the fuel filter. The fuel control has a fuel metering section and a computing section. The metering section selects the rate of flow to the combustion chambers, based on information received from the computing sections. The metering section has a metering valve and a pressure regulating valve. The pressure regulating valve maintains a constant pressure across the main metering valve by bypassing excess fuel back to the engine fuel

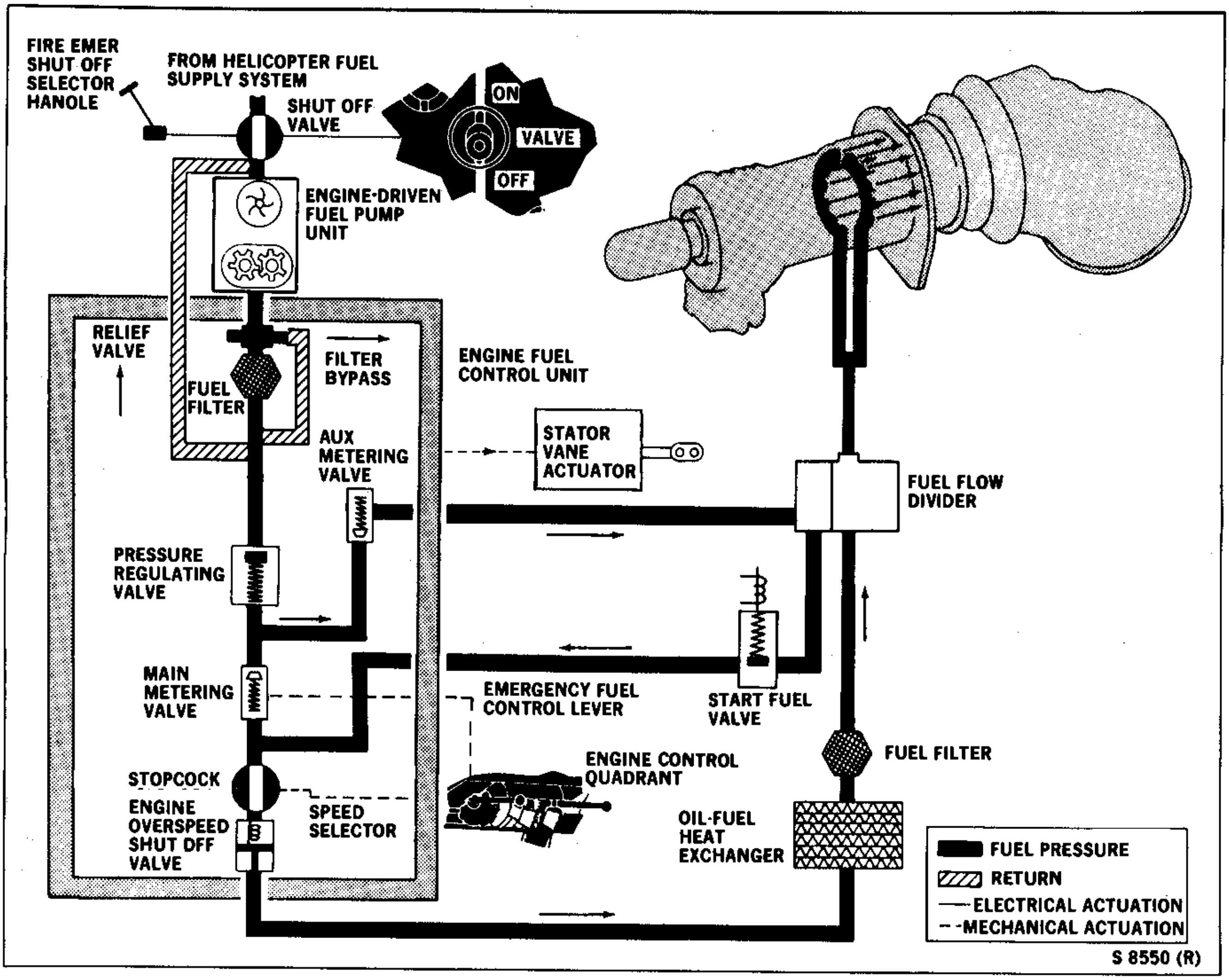

Figure 1-3. Engine Fuel System

pump inlet. The metering valve is positioned in response to various internal operating signals, and meters fuel to the engine as a function of these integrated signals. The engine fuel control unit prevents compressor stall, turbine overtemperature, rich or lean blowouts, governs gas generator idle and maximum speeds, and schedules inlet guide and stator vane positions to provide optimum compressor performance.

SPEED SELECTORS (ENGINE SPEED SELECTOR LEVERS)

Two speed selectors, marked NUMBER 1 ENGINE and NUMBER 2 ENGINE, are located on the overhead engine control quadrant (figures 1-4 and FO-3). Marked positions on the overhead quadrant are SHUT-OFF, GRD IDLE, MIN GOV, and 100% SPEED. The speed selectors are connected directly to the fuel stopcock and indirectly to the fuel metering valve in the fuel control unit. When the speed selectors are in the SHUT-OFF position, fuel flow to the fuel nozzles is stopped by means of a stopcock that prevents fuel from entering the combustion chambers. The stopcock is open whenever the speed selector is 6° or more from the SHUT-OFF position and is closed when the speed selector is 3° or less from the SHUT-OFF position. The GRD IDLE position schedules fuel flow to produce approximately 56% N_g. Gas generator idle speed will vary with inlet air temperature. A limit stop at GRD IDLE prevents inadvertent retarding of the speed selectors below the idle speed of the engines. The speed selectors may be retarded from the limit stop by exerting a downward and rearward pressure on the speed selectors. The MIN GOV position of the speed selector is the point where the governing range of the power turbine is entered and is approximately 87% N_f. When the speed selector is at the full forward position, the engine is producing maximum power turbine speed. Engine speed trim switches are installed on the collective stick grip to provide accurate speed changes and engine synchronization. With the speed selectors in the governing range, any force tending to slow the rotor system, such as increases in collective pitch, will be sensed by the fuel control unit,, which will attempt to maintain constant Nf/Nr by increasing power.

ENGINE SPEED TRIM SWITCHES

The trim switches, on each collective lever grip (figure 1-5), are used to make adjustments to power turbine speed and for engine synchronization. The switches, marked ENG TRIM, 1 and 2, + (plus) - (minus), provide electrical power to actuators in the overhead control quadrant, which are connected to the speed selectors. The pilot's engine speed trim switches will override the copilot's. Moving the ENG TRIM switches forward will cause increases in power turbine speed, and moving the switches aft will cause decreases in power turbine speed. When the desired power turbine speed is attained, the switches are released and will return to the spring-loaded center position. The switches are capable of advancing the speed selectors from shutoff to the ground idle detent and, if manually removed from the detent, to the full forward position. The switches are capable of retarding the speed selectors from full forward to 93% ± 1% Nf. The ENG TRIM switches receive electrical power from the dc primary bus through circuit breakers, under the general heading ENGINE (SPEED TRIM, 1-ENG-2), located on the center overhead dc circuit breaker panel.

EMERGENCY FUEL CONTROL LEVERS

Two emergency fuel control levers, one for each engine, marked EMER FUEL CONTROL, are located on each side of the engine control quadrant (figure 1-4). The emergency fuel control levers operate independently and are used in event of fuel control unit malfunction. Each emergency fuel control lever has positive stops, marked OPEN and CLOSE, and is connected directly by a flexible cable and linkage to the main metering valve in each engine fuel control unit. The primary function of the emergency fuel control lever is to manually override the automatic features of the fuel control, and must be used with extreme caution as it has direct control of fuel flow. Misuse can cause engine overspeed or overtemperature. The initial position of the fuel metering valve is dependent upon the automatic features of the fuel control, as established by the setting of the speed selector. The emergency fuel control lever is mechanically connected to a cam within the fuel control. This cam, when actuated by advancing

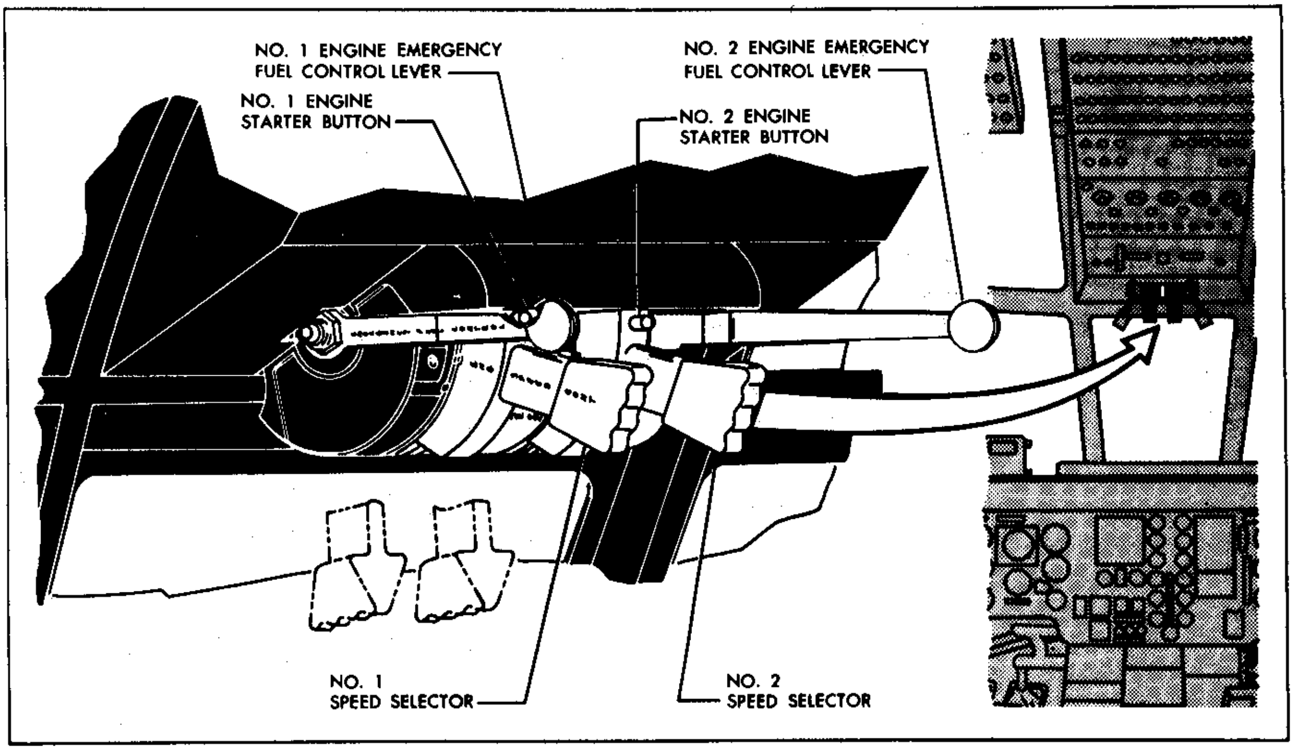

Figure 1-4. Engine Controls

the lever, contacts the fuel metering valve. Once contact is established, further advancement of the emergency fuel control will manually control fuel flow, which in turn regulates engine power output. The emergency fuel control is unable to reduce the position of the metering valve below that called for by the speed selectors. Control below this point will depend upon the type of malfunction encountered. In all instances of emergency fuel control operation, it must be remembered that the speed selectors must not be retarded beyond the GRD IDLE position. The fuel stopcock is located downstream of the metering valve and is actuated by the speed selectors. Placing the speed selector in the SHUT-OFF position will stop engine fuel flow regardless of emergency fuel control lever position.

CAUTION

At high power settings, considerable dead band travel will normally be encountered before the emergency fuel control lever becomes effective, indicated by a slight restriction in control movement. When this is felt, the control will be very sensitive and care should be taken not to exceed T_5 and N_g limits.

STARTER SYSTEM

The starting system consists of a starter, starter relay, start bleed valve, start fuel valve, starter button, mode selector switch, and abort switch. The system operates on 28 volts dc from the primary bus (24 volts if only battery used) and is protected by circuit breakers, marked STARTER 1 ENG 2, located on the overhead circuit breaker panel. The engine starting system (figure 1-6) has two modes of operation: normal and manual. The normal mode provides a completely automatic start, which includes automatic starter drop out after engine light-off. The manual mode provides an alternate means of starting when using external electrical power or the battery. In this mode, the automatic starter dropout feature is bypassed. In either mode, the start bleed valve operates automatically and the auxiliary start fuel shutoff valve may be operated by the pilot. The starter has a duty cycle limited to 30 seconds continuous cranking, a minimum cooling period of 3 minutes between start attempts, and a maximum of three start attempts in any 30-minute period. Before the starter can be energized, APU, external, or battery power is required. During start, as the engine speed selector lever is advanced to the GRD IDLE position, the stopcock opens and allows fuel to pass through the flow divider and to enter the number one, low pressure, manifold to the nozzles where it is mixed with compressor discharge air. As the fuel-air mixture leaves the nozzles, it is ignited by the two

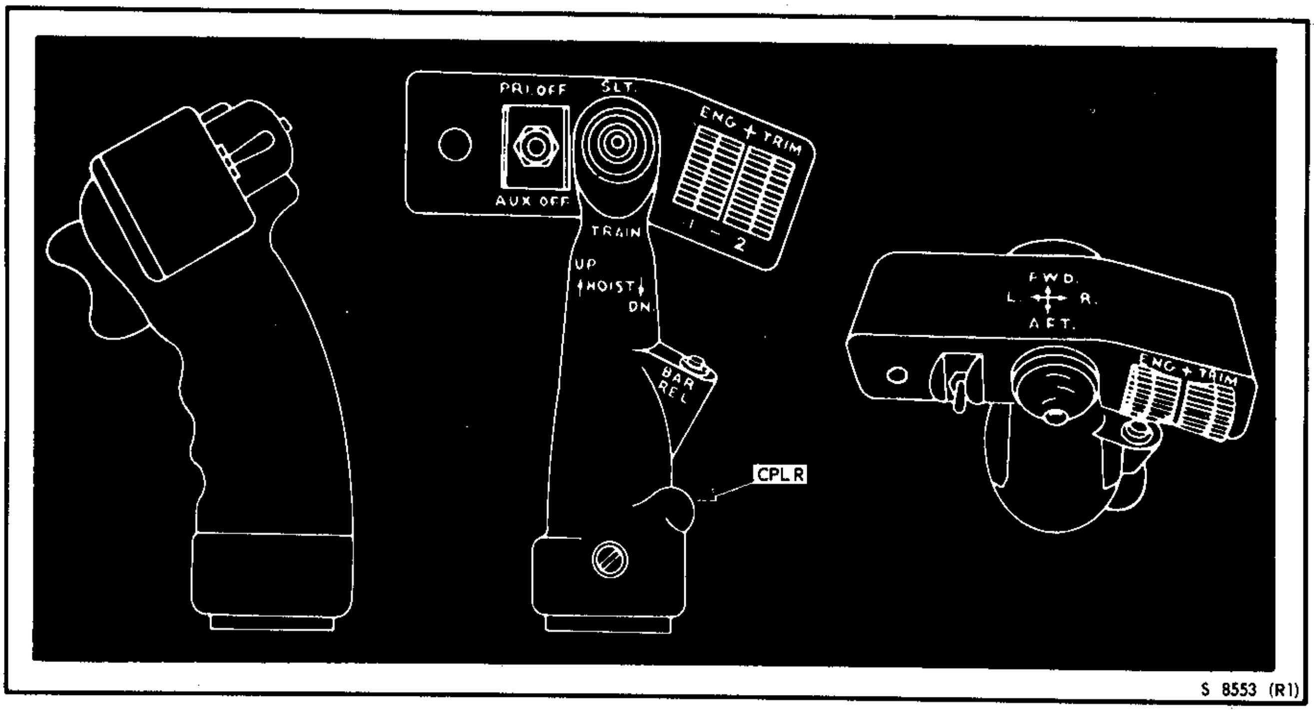

Figure 1-5. Collective Pitch Lever Grip

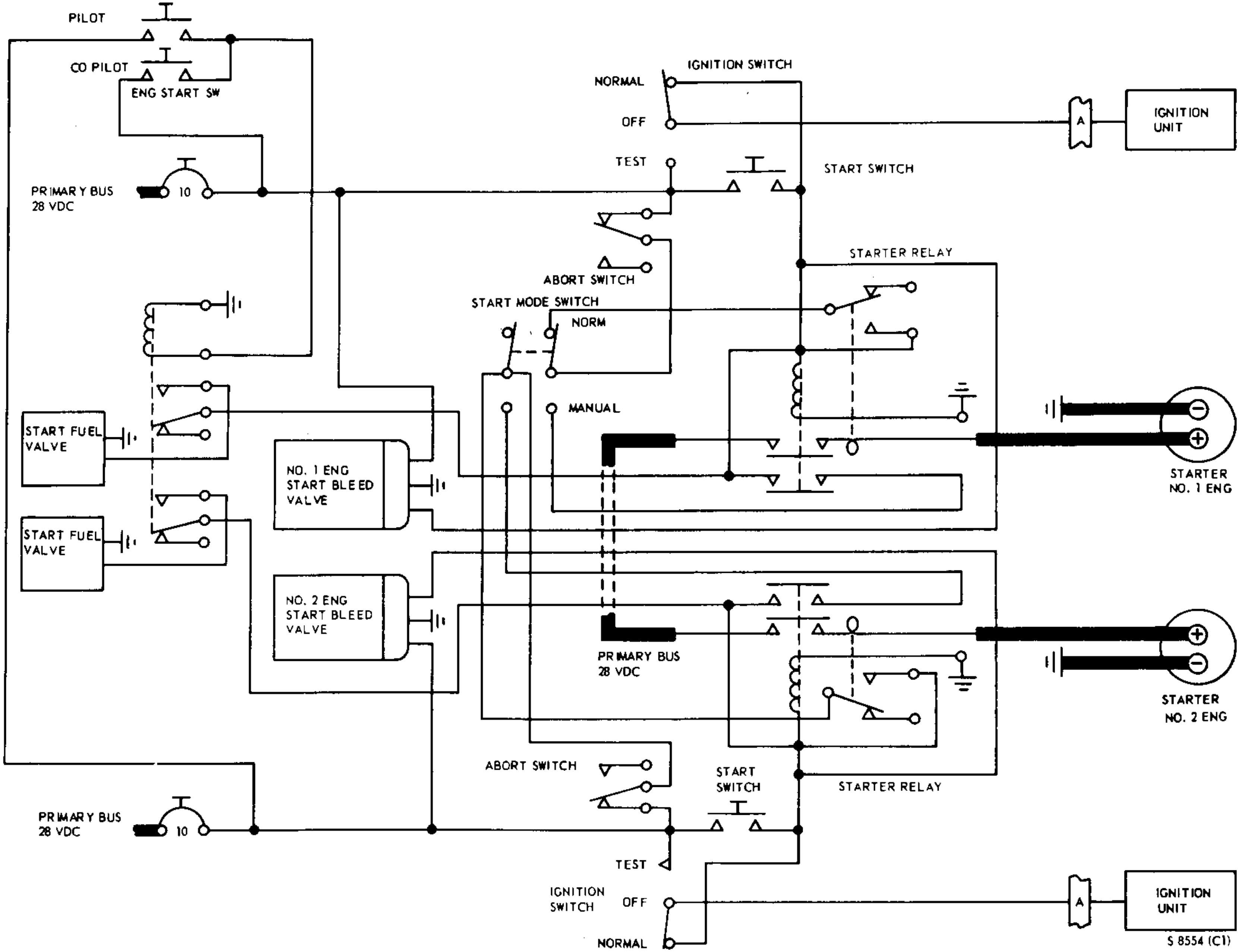

Figure 1-6. Engine Starting System

igniter plugs in the combustion chamber and enters a sustained combustion process.

Starter Dropout

Starter operation and dropout may be monitored by noting the magnetic compass heading prior to engine start. When the starter is energized, the compass will swing to a new heading. When normal start circuit is used, the starter will automatically drop out when starter amperage falls below 100 ± 15 amperes (45% to 53% N_g). The compass should then swing back to its original heading signifying the starter has dropped out. When the manual start circuit is used, the starter will only drop out when the abort switch is actuated.

Start Bleed Valve

The start bleed valves, located on each engine, operate automatically during the start cycle and require no specific pilot action. The function of the valve is to raise the compressor stall line during the start cycle in order to increase the reliability of the start system. This is accomplished by bleeding approximately 6.7% of the compressor discharge air flow at ground idle. The bleed valve remains fully closed during all regimes of engine operation, except during starting.

Starter Buttons

A starter button is located on each speed selector lever. The starter is energized by holding the speed selector lever in the SHUT-OFF position and momentarily depressing the starter button. This energizes the starter relay and completes the circuit to the starter. When using normal starting mode, after engine lite-off and the electrical power load to starter decreases, the starter relay will automatically drop out, deenergizing the starter. When using the manual mode, the respective speed selector must be pulled down to actuate the abort switch, which in turn drops out the starter.

Starter Abort Switch

A starter abort switch, located in each speed selector lever, is actuated by pulling down on the speed selector lever. This action breaks electrical circuit continuity to the ignition system and the starter relay.

Mode Selector Switch

The mode selector switch, with marked positions MANUAL and NORMAL under the general heading START MODE, is located on the overhead control panel. When the switch is placed in the NORMAL position, the automatic dropout function of the starter relay is energized to allow the starter to remain energized to 45% to 53% N_g. When the switch is placed in the MANUAL position, the automatic dropout feature of the starter relay is bypassed to allow the starter to remain energized until the abort switch is actuated manually. The switch operates on 28 volts dc from the primary bus.

Start Fuel Valves

Two start fuel valves, one for each engine and located in the engine compartment, are installed between the engine fuel control and flow divider. When the valve is actuated during the start cycle of either engine, the flow of auxiliary starting fuel is blocked. This blockage decreases the total amount of fuel flow during starting, thus diminishing the possibility of an overtemperature condition due to excessive fuel flow. The valves operate on 28 volts dc from the primary bus and are protected by the main starting circuit breakers.

Start Fuel Valve Switch The pushbutton type start fuel valve switch, marked ENG ST, is located on each cyclic stick grip. In addition to depressing the switch, the starter relay for the engine to be started must be closed before the valve will operate. Either the pilot or copilot's switch will control the operation of both valves. The switch operates on 28 volts dc from the primary bus.

IGNITION SYSTEM

Each engine ignition system consists of a capacitor-discharge ignition unit, two ignitor plugs, and a control circuit. The system provides ignition for starting only; during engine operation, the flame in the combustion chamber is self-sustaining. The ignition system, mounted on the engine, is controlled by a three-position switch mounted on the overhead switch panel. When the switch is in the NORM position, the ignition unit operates in conjunction with the starter. The ignition system is de-energized when the starter is de-energized. The ignition system operates on current from the dc primary bus through the starter control system.

Ignition Switches

Two ignition switches, one for each engine and located on the overhead switch panel (figure FO-3), are marked IGNITION, 1 ENG, 2. Each switch has marked positions TEST, OFF, and NORM, and are normally in the NORM position. When the switch is in the NORM position with the starter engaged, the ignition unit is energized. Holding the switch in the spring-loaded TEST position, energizes the ignition unit only. The TEST position is used for ground operation only, without the starter, to test the ignition circuit. A clicking can be heard when the switch is placed in TEST position. When the switch is in the OFF position, the ignition unit is de-energized. The engine may be motored by using the starter without ignition. The speed selector must be in the SHUT-OFF position before the starter and ignition systems can be energized.

ENGINE INSTRUMENTS

Torquemeters

Two torquemeters (1 and 45, figure FO-4), one for the pilot and one for the copilot, are on the instrument panel. Each dual-pointer indicator, marked PERCENT TORQUE, contains two pointers, marked 1 and 2, which indicate input torque to the main transmission in percent of maximum engine power output of each engine. The electrically actuated torquemeter dials, calibrated in percent torque, are graduated in units of 5% from 0% to 150%. The torquemeters operate on 26 volts ac and are protected by circuit breakers, marked 1 ENG TORQUE SENSOR, on the copilot's ac circuit breaker panel, and 2 ENG TORQUE SENSOR, on the pilot's ac circuit breaker panel. The torquemeters indicate the amount of torque being applied to the main gear box by the engines. The torque sensing cells are in the main gear box and are hydromechanical in nature, sensing any shift in the helical gear at the input from each engine. Oil pressures within the cells are sensed by pressure transmitters and transmitted electrically to the torquemeters.

Engine Gas Generator (N_g) Tachometers

Two engine gas generator tachometers (22 and 33, figure FO-4), one for each engine, are located on the instrument panel and indicate the speed of the gas generator in percent rpm. Each tachometer has two dials and pointers. The outer dial and pointer indicates gas generator speed from 0% to 100% in increments of 2%. The small vernier dial and pointer, located in the upper left-hand position of the tachometer, indicates gas generator speed from 0 to 10, in increments of 1%. The gas generator tachometer generator is driven by the engine oil pump shaft. The electrical power produced by the gas generator tachometer generator is proportional to gas generator rpm (100% N_g = 26300 gas generator rpm).

N_f and N_r Triple Tachometers

Two triple tachometers (2 and 46, figure FO-4), one for the pilot and one for the copilot, located on the instrument panel, each contain three pointers. The pointers marked 1 and 2 indicate the power turbine speed (N_f) of the No. 1 and 2 engines, respectively, the pointer marked R indicates the main rotor rpm (N_r). The engine tachometers are powered by their own tachometer generators, which are driven by the power turbine through a flex cable, which is routed to the fuel control on which they are mounted. The main rotor tachometer is powered by its own tachometer generator, located on the accessory section of the gear box, and driven by the accessory gears. The tachometers are read in percent rpm (103% N_f = 19,500 power turbine rpm and 103% N_r = 210 rotor rpm).

Power Turbine Inlet Temperature (T_5) Indicators

Two power turbine inlet temperature indicators (23 and 34, figure FO-4), marked EXH. TEMP, are located on the instrument panel. The indicators are graduated in degrees Celsius and operate from thermocouples, located forward of the power turbine in the second-stage turbine casing, on each engine. The maximum power turbine inlet temperature is indirectly controlled by the gas generator speed adjustment of the fuel control.

Engine Oil Pressure Indicators

Two oil pressure indicators (25 and 36, figure FO-4), one for each engine, are on the instrument panel. The indicators are powered by 26 volts ac from separate autotransformers and are protected by circuit breakers marked OIL PRESS 1 ENG, on the copilot's ac circuit breaker panel, and OIL PRESS 2 ENG on the pilot's ac circuit breaker panel. Pressure is indicated in psi.

Engine Oil Temperature Indicators

Two engine oil temperature indicators (26 and 37, figure FO-4), one for each engine, are located on the instrument panel. The engine oil temperature bulb, located on each oil inlet line, transmits indications to the respective temperature indicators. The indicators are powered from the dc primary bus and are protected by circuit breakers, marked OIL TEMP, 1-ENG-2, on the overhead dc circuit breaker panel. Temperature is indicated in degrees Celsius.

Fuel Flow Indicators

Two fuel flow indicators (24 and 35, figure FO-4), calibrated in pounds per hour, are located on the instrument panel. The fuel flow indicators provide indication of the fuel consumption of the engines and operate on 115-volt ac electrical power from respective ac primary buses. They are protected by circuit breakers, marked FLOW, under the general heading FUEL. The circuit breaker for the No. 1 engine is located on the copilot's ac circuit breaker panel, and for the No. 2 engine on the pilot's ac circuit breaker panel.

ROTOR SYSTEMS

The rotor systems consist of a single main rotor and an anti-torque tail rotor. Both systems are driven by the two engines through the transmission system and are controlled by the flight controls.

MAIN ROTOR SYSTEM

The main rotor system consists of the main rotor head assembly and the rotor blades. The head assembly, mounted directly above the main gear box, consists of a hub assembly and a swashplate assembly. The hub assembly, consisting of five sleeve-spindle assemblies and five hydraulic dampers clamped between two parallel plates, is splined to the main rotor drive shaft. The root ends of the five rotor blades are attached to the sleeve-spindle assemblies, which permit each blade to flap vertically, hunt horizontally, and rotate about their span-wise axis. Anti-flapping restrainers limit the upward movement of the blades and droop stops limit the downward movement of the blades. Both are in operation when the blades are stopped or turning at low speed. When speed is increased to approximately 25% (50 rpm) rotor speed, centrifugal force automatically releases the anti-flapping restrainers. The droop stops release at approximately 75% (152 rpm) rotor speed. The hydraulic dampers minimize hunting movement of the blades about the vertical hinges as they rotate, prevent shock to the blades when the rotor is started or stopped, and aid in the prevention of ground resonance. The five all-metal main rotor blades are of the pressurized spar type. The 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. Each blade consists of a hollow extruded aluminum spar pressurized with nitrogen, 25 aluminum blade pockets, a tip cap, an aluminum root cap, a steel cuff, a pressure (IBIS) indicator, an air valve, and an abrasion strip. Vent holes on the underside of each pocket prevent accumulation of moisture inside the blade. Each blade is balanced statically and dynamically within tolerances that permit individual replacement of the blades. In addition, a pretrack number is stenciled on each blade to eliminate the necessity for blade tracking. Balancing and the assignment of a pretrack number is done during manufacture or overhaul. The swashplate assembly consists of an upper (rotating) swashplate, which is driven by the rotor hub, and a lower (stationary) swashplate, which is secured by a scissors assembly to the main gear box to prevent rotation. Both swashplates are mounted on a ball-ring and socket assembly, which keeps them parallel at all times, but allows them to be tilted, raised, or lowered simultaneously by components of the main rotor flight control system, which connect to arms on the lower (stationary) swashplate. Cyclic or collective pitch changes, introduced at the stationary swashplate, are transmitted to the blades by linkage on the rotating swashplate.

Bifilar Absorber

The main rotor system is equipped with a bifilar absorber assembly to reduce fatigue stress and improve the overall vibration comfort level throughout the helicopter. The bifilar absorber assembly, secured to the main rotor hub, consists of a five-pointed, star-shaped, aluminum forging with a seventeen pound weight attached to each star point. Each weight is enclosed by a fairing to reduce drag.

IBIS (In-Flight Blade Inspection System) Indicators

The In-Flight Blade Inspection System (IBIS) (figure 1-7) visibly indicates in the cockpit that the pressure in one or more main rotor blade spars has dropped below the allowable limit.

The IBIS indicator, located on the back wall of the spar of each main blade, contains a small radioactive source (100 micro curies strontium 90) which is completely shielded and emits no radiation when the rotor blade spar is at normal pressure. When the pressure in the rotor blade spar drops below 6.1(±0.4)psi, the indicator will activate, causing the radioactive source to move to an unshielded position, thereby emitting beta radiation. The detector assembly, located aft of the main rotor shaft under the transmission cowling, detects the beta radiation and sends a signal to the signal processor located on the forward electronics rack immediately aft of the copilot. The signal processor causes the BLADE PRESS light on the caution advisory panel to illuminate, indicating a loss of pressure.

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in one or more of the blade spars. Loss of pressure in the blade spar is also indicated by the IBIS indicator located in the back wall of the spar of each main blade. The indicator, compensated for temperature changes, compares a reference pressure built into the indicator with the pressure in the blade spar. When the pressure in the blade spar is within the required service limits, two yellow stripes show in the indicator. If the pressure in the blade spar drops below the minimum permissible service pressure, the indicator will be activated and will show two red stripes. Loss of 115 volt ac power, failure of the detector, and/or failure of the signal processor will cause the BLADE PRESS caution light to illuminate. The system receives electrical power from the No. 1 ac primary bus and is protected by a circuit breaker on the copilot's overhead circuit breaker panel marked IBIS, under the general heading NO 1 AC PRI. The BLADE PRESS caution light is powered by the dc primary bus, and is protected by a circuit breaker on the copilot's overhead circuit breaker panel marked IBIS, under the general heading DC PRI.

WARNING

When red is visible in the indicator, the cause of the red indication shall be determined before accepting the helicopter for flight.

NOTE

A protective plastic cover, designed to eliminate radiation leakage, is carried on board each aircraft and shall be placed over any red IBIS Indicator to preclude radiation leakage.

TAIL ROTOR

The tail rotor consists of the tail rotor assembly and tail rotor blades. The tail rotor assembly, mounted at the upper end of the pylon, consists of a tail rotor hub and pitch-changing mechanism. The splined hub is supported and driven by the horizontal output shaft of the tail gear box. The five tail rotor blades are attached to the tail rotor hub in such a way that they are free to flap and rotate about their spanwise axis for pitch variation. The blade pitch-changing mechanism transmits tail rotor control pedal movements to the tail

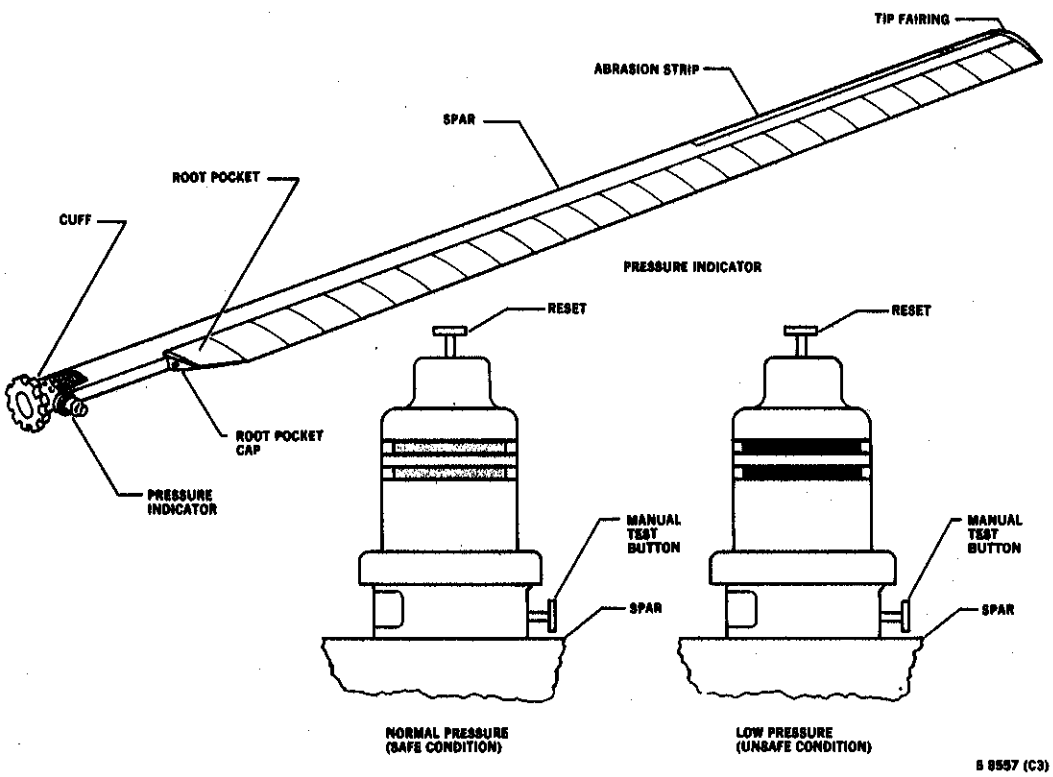

Figure 1-7. IBIS Indicator

rotor blades through the horizontal output shaft of the tail gear box.

ROTOR BRAKE

A **hydraulically-actuated** rotor brake (figure 1-8), mounted on a brake shaft forward of the main gear box, stops the rotation of the rotor system and prevents its rotation, when the helicopter is parked. The rotor brake system consists of a hydraulic cylinder and lever, pressure gage, hydraulic brake cylinders, rotor brake reservoir, and a brake disc. The rotor brake hydraulic cylinder and lever, located on the pilot's compartment ceiling, operates independently from the hydraulic systems. A spring-loaded accumulator, connected to the rotor brake hydraulic lines at the forward end of the transmission compartment, assures continuous hydraulic pressure when the rotor brake lever is applied. The rotor brake hydraulic cylinder is gravity-fed with hydraulic fluid from the rotor brake hydraulic reservoir, located on the right aft side of the main gear box. When filled to the FULL mark, the reservoir contains approximately 3.2 ounces of fluid. The hydraulic brake cylinder is located on supports attached to the main gear box. The brake disc is positioned on the main input shaft of the main gear box.

Rotor Brake Cylinder and Lever

A rotor brake lever (figure 1-8) is connected directly to the rotor brake hydraulic cylinder, on the pilot's compartment ceiling to the right and forward of the overhead switch panel. The rotor brake is applied by pulling down and pushing forward as indicated on the decal aft of the lever on the upper structure. The decal is marked TO ENGAGE ROTOR BRAKE PUSH LEVER FORWARD. The decal also has an arrow pointing forward. A spring-loaded lock, located at the forward outboard side of the cylinder, automatically locks the brake lever in the applied (forward) position if the pilot places the small handle on the horizontal, forward position. The lockpin must be pulled out in order to allow rotor brake release. The lockpin may be rendered inoperative by rotating it until it remains in the OUT position.

Rotor Brake Pressure Gage

A hydraulically-actuated rotor brake pressure gage is located to the rear of the rotor brake lever (see figure 1-8), on the pilot's compartment ceiling. The reading is indicated by the pointer and indicates PSI. A decal, marked ROTOR BRAKE PRESSURE, ACTUATING RANGE 350-500 PSI., ENGINE START 320 PSI MIN. PARKED POS. RANGE 250-600 PSI., is located adjacent to the rotor brake pressure gage. To ensure that the rotors will not turn with both engines in GRD IDLE, a minimum of 320 psi is required before starting engines. For parking, 250-600 psi is the normal pressure range.

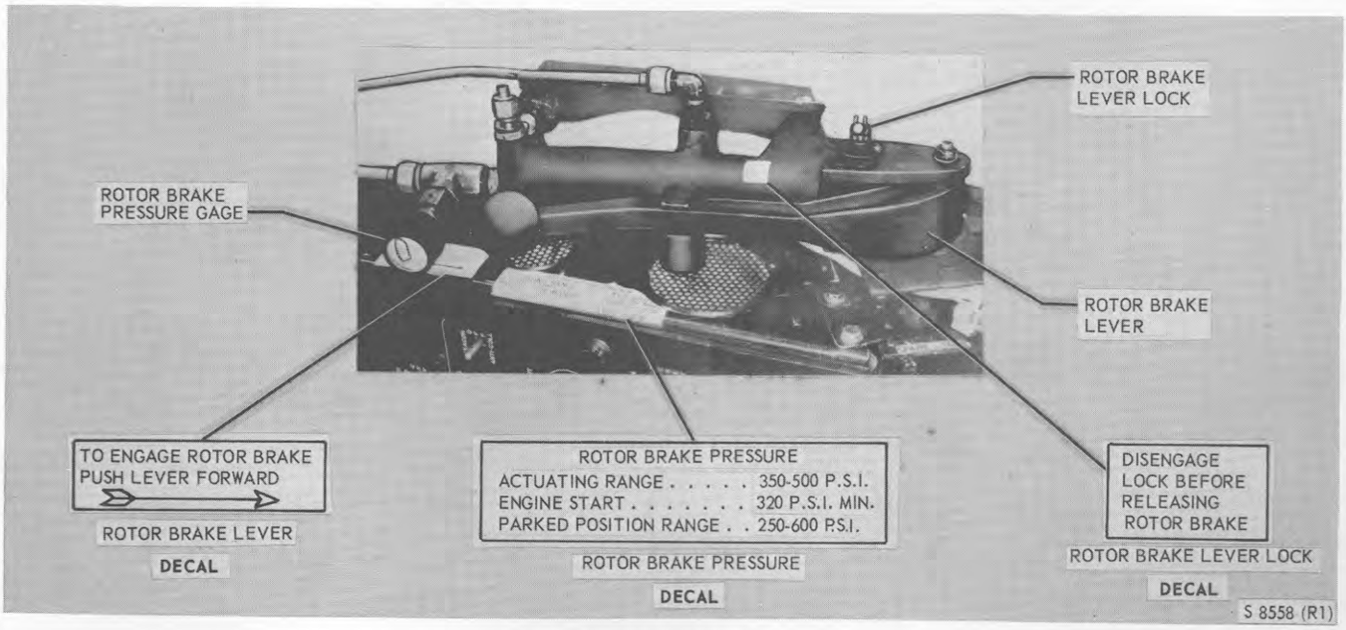

Figure 1-8. Rotor Brake