Bell Helicopter

A Textron Company

407 Pilot Ground and Flight Procedures

8-31-2002

Training Academy

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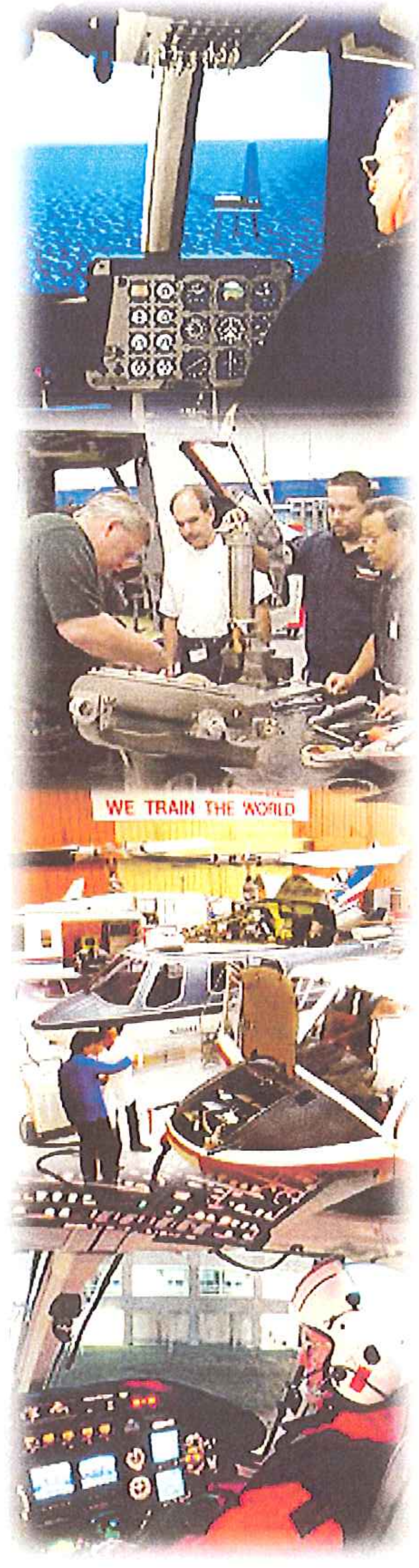

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INFORMATION PAGE

Instructor Information

Name:________________________________ Email:__________________________

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Student Information

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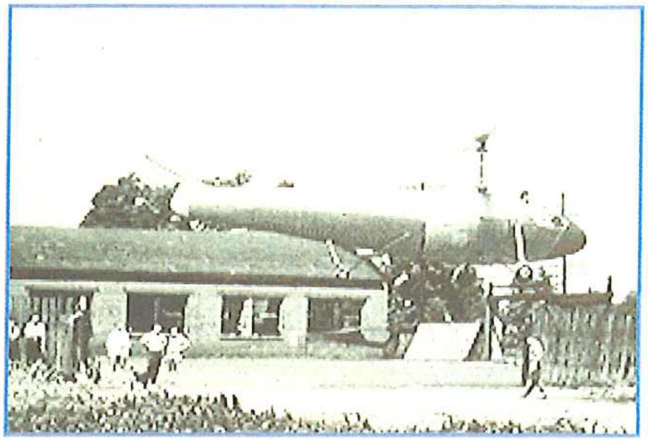
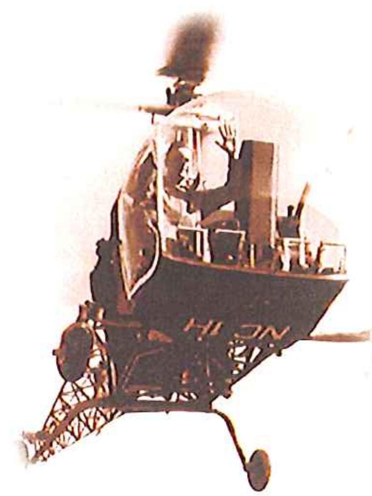
BELL HELICOPTER HISTORY



Lawrence D. Bell, a man referred to as "America's most seasoned dreamer", founded Bell Aircraft Corporation, the predecessor of Bell Helicopter Textron July 10, 1935 in Buffalo, New York. Bell Aircraft Corporation created a Helicopter Division, which moved to Fort Worth, Texas in 1951 and became Bell Helicopter Corporation, a wholly owned subsidiary of Bell Aircraft Corporation.

In 1960, Textron of Providence, Rhode Island bought various Bell Aircraft properties including the Helicopter operation. Textron changed the name of the helicopter operation to Bell Helicopter Company, and established itself as Textron's largest division. In 1976 the name changed to "Bell Helicopter Textron".

In 1946 pioneers like Floyd Carlson, Joe Mashman, Art Young and Dick Stansbury were experimental pilots developing the first Bell Helicopter models in Buffalo, New York. In 1947 they conducted flight instruction in NC-1H a Model 47; the first helicopter to receive CAA certification. In those days the Buffalo operation was essentially the only place where a pilot could get flight instruction in helicopters. The first Army helicopter pilots took their training from Bell Helicopter at this Gardenville, N.Y. location.



When Bell Aircraft moved the helicopter operation to Fort Worth, Texas, all of the initial operations, including flight and maintenance training in the Model 47 took place at a leased facility in Saginaw, Texas just north of Fort Worth. This original plant, which was used for a variety of design, production, testing and training operations were commonly called the Globe Plant. This was a Naval facility that was acquired to manufacture the Cessna "Bamboo Bombers" late in the WW II era

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HISTORY CONTINUED

After the war this flying field was leased to Bell Helicopter. For history buffs this was the original Globe "Swift" Airplane manufacturing facility. Pilot and Maintenance training continued at the Globe Plant until 1970 when a new classroom and maintenance instruction hangar was built in Hurst just southwest of the main plant. This is the current location of the Bell Helicopter Training Academy and Delivery Center. Initially, Pilot and Mechanic Training was limited to the Model 47 and the 206 JetRanger. This new Bell training facility offered hands on pilot and maintainer training using real aircraft in a setting considered to be state of the art training. This modern facility hosted training conducted by a staff of 20 people including two Instructor Pilots. The Model 206A "JetRanger" was added to the commercial product line in 1967, and as other models were produced the Training Academy staff and facility continued to grow.

In 1996, the Training Academy opened an 18,000 square-foot wing, providing much-needed additional office and classroom space to handle the heavy customer load. From a mere trickle when Bell began training in Buffalo in the 40's, the Training Academy saw its 80,000th customer complete training in December of 2000. Traditional mechanic and pilot training has been the primary product of the Training Academy; but it also has the capability of producing state-of-the-art Computer Based Training (CBT). Developed primarily to meet the needs of the U.S. Military for OH-58D and V-22 pilot and mechanic training, this CBT is also being applied to commercial training programs.

The Bell Helicopter Training Academy now has a staff and offices of 150 people; 100,000 square feet of maintenance training hangar space; 8 helicopters; 25 classrooms and laboratories, 3 Flight Training Devices; and the desire to deliver the highest quality products and services to the helicopter industry. It has been the quality of training and the genuine interest in the customer that has built the Bell Helicopter Training Academy into the finest helicopter training facility in the world; which created the motto "We Train the World".

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Bell Helicopter

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General Description

Training Academy

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GENERAL DESCRIPTION

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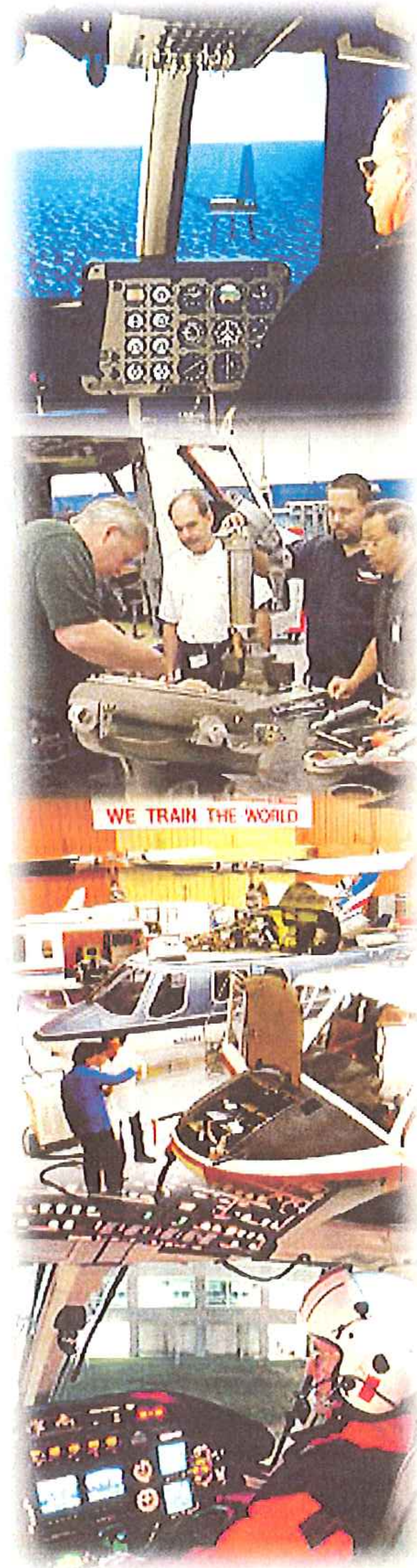

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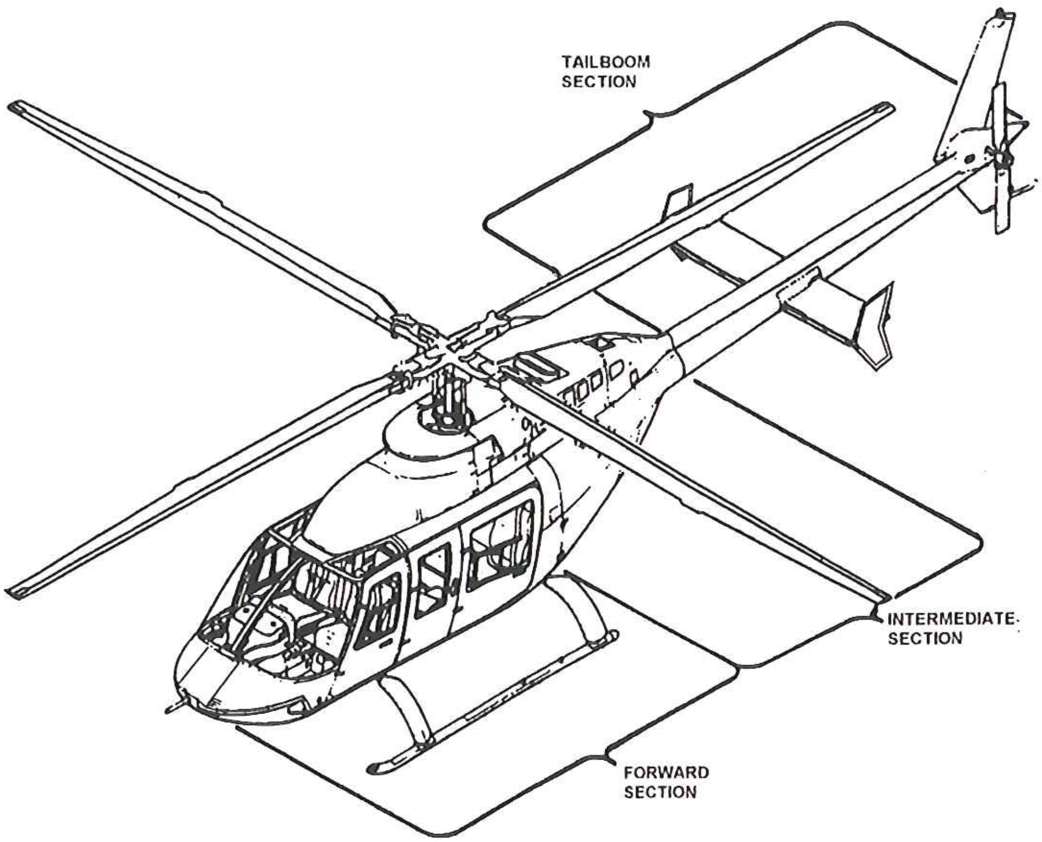
GENERAL DESCRIPTION

Helicopter Description

The Bell Model 407 is a single engine, seven place helicopter designed to takeoff and land on any reasonably level terrain. Standard configuration provides for one pilot and six passengers.

The fuselage consists of three main sections: the forward section, the intermediate section, and the tailboom section. The forward section utilizes aluminum honeycomb and carbon graphite structure and provides the major load carrying elements of the forward cabin. The intermediate section is a semi-monocoque structure that uses bulkheads, longerons and carbon fiber composite side skins. The tailboom is an aluminum monocoque construction that transmits all stresses through its external skins.



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GENERAL DESCRIPTION

Airframe

The airframe utilizes aluminum honeycomb carbon graphite structure for the forward section, a semi-monocoque structure with longerons, and carbon fiber composite side skins for the intermediate section, and an aluminum monocoque tail boom.

The basic helicopter landing gear is the low skid type. Various landing gear configurations including high skid, or emergency flotation gear are available as kits.

The 407 uses skid type landing gear secured to the fuselage by means of three-point attachment. Each skid tube is fitted with a tow fitting, replaceable skid shoes, and two eyebolt fittings for the mounting of ground handling gear.

Cowlings and fairings enclose the various roof and tail boom mounted assemblies. Access doors and inspection windows (cutouts), are provided for preflight and inspections. Cowlings are manufactured from composite or aluminum materials and are readily removable for maintenance access.

Passenger seat cushions can be removed or rearranged in order to permit cargo to be carried internally. The aft cabin doors may be removed to permit carrying wide loads. A baggage compartment is located in the intermediate section beneath the engine compartment. A number of available kits (aircraft supplements) may be used to configure the aircraft for mission specific needs.

Engine

The Bell Model 407 is powered by a Rolls Royce turboshaft engine, model 250-C47B, that provides 674 Shaft Horsepower (SHP) for takeoff and 630 SHP for maximum continuous operation. An electronic Full Authority Digital Electronic Control (FADEC) is integrated to provide additional reliability during engine starts, improved fuel control during flight, and continuous in-flight systems monitoring.

The use of this light weight gas turbine engine, a high strength to low weight ratio bonded honeycomb constructed airframe, simplified dynamic components, and the soft mounted pylon isolation system to dampen main rotor vibrations, has produced a precision balance between performance, dependability, and capability.

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GENERAL DESCRIPTION

Rotor System

The main rotor is a four bladed soft - in - plane, flex beam type hub with individually interchangeable blades. The main rotor blades and rotor hub utilize composite construction. The pitch change bearings and lead - lag dampers utilize elastomeric technology that require no maintenance and have benign failure modes. Main rotor spindles, the upper and lower hub plate, and pitch change horns are constructed of aluminum forgings for strength and reduced weight.

The tail rotor is a two bladed teetering rotor that provides directional control. It incorporates a semirigid, delta hinged design. The blades are composite and are mounted to a steel yoke.

Fuel

The basic aircraft has a usable fuel capacity of 127.8 U.S. Gallons distributed within two cells. A 20 U.S. Gallon extended range tank is also available as optional equipment.

Crew Compartment

The crew compartment provides for one or two pilot operation, with the pilot station located in the right seat. An instrument panel is mounted on a central pedestal forward of the pilot and copilot seats. An overhead electrical panel places the majority of switches and circuit breakers within easy reach of the pilot.

Crew seats are covered with a flame retardant fabric. The seat belts are designed with inertia reel equipped shoulder restraints.

All aircraft doors are hinged to the front and are equipped with internal and external, easy to operate automobile style door handles and locks. All windows are manufactured of gray tinted acrylic plastic.

Passenger Compartment

The rear passenger compartment incorporates a five place seating configuration which seats three passengers looking forward and two facing rearward. Passenger seats are covered with flame retardant fabric and several internal seating choices are available. The seat belts are designed with inertia reel shoulder restraints.

The large passenger doors are equipped with internal and external, easy to operate, automobile style door handles. An additional door (litter door) located on the aircraft left side provides for an increased opening to load large items including patients on stretchers. The windows are manufactured of gray tinted acrylic plastic.

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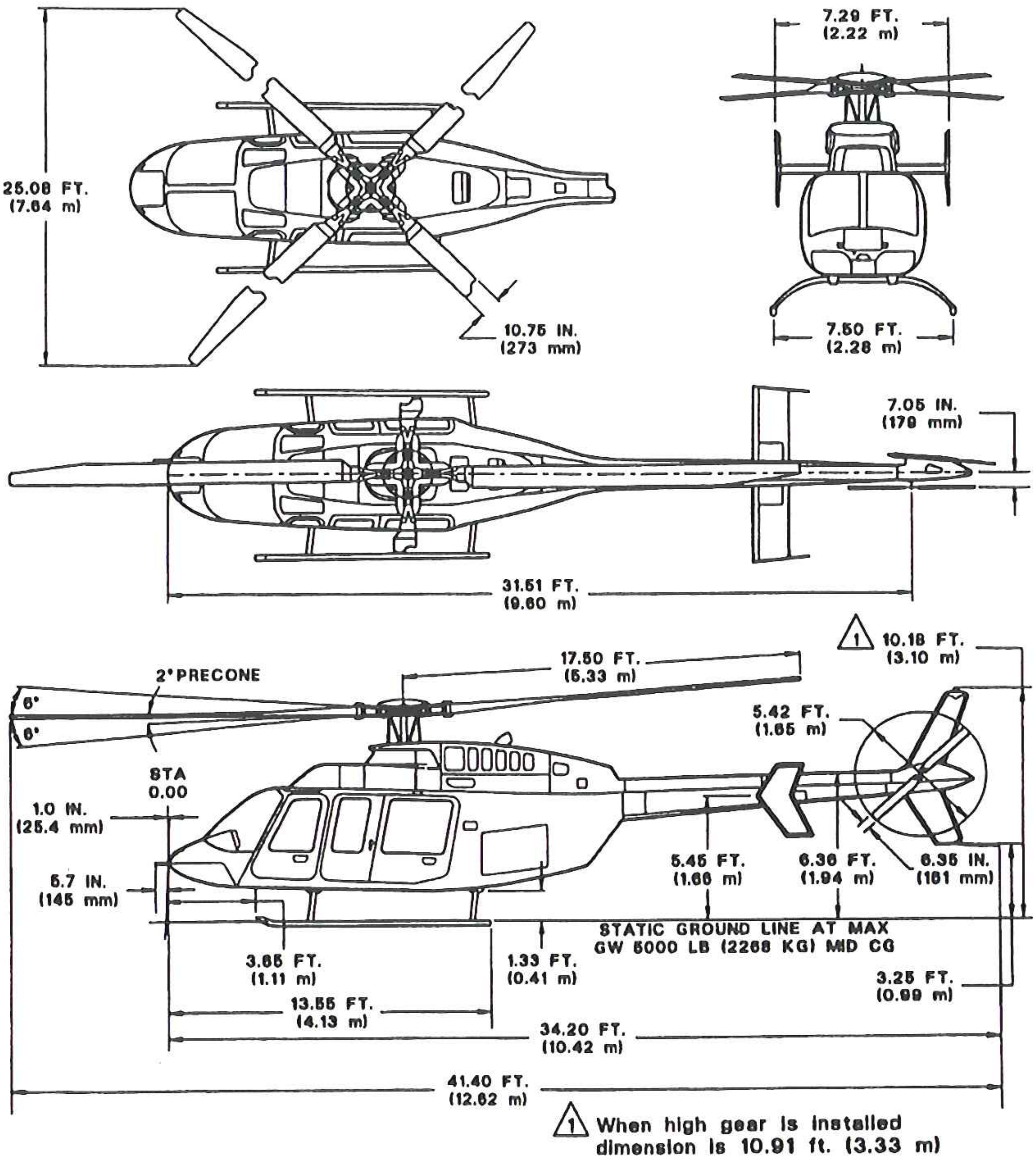

TYPICAL

407-MD-1-1

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GENERAL DESCRIPTION

General Data Table

Engine

Model number	250-C47B
Manufacturer	Rolls Royce

Torque / Horsepower

Maximum continuous power	93.5% Torque/630 Shaft Horsepower
Takeoff power (5 minute limit)	100% Torque/674 Shaft Horsepower

Airspeed Limitations (Indicated)

Maximum airspeed V_{NE}	140 Knots
Maximum airspeed for Autorotation	100 Knots
Maximum airspeed at 93.5 to 100%Torque	100 Knots
Maximum airspeed with doors off (any combination removed)	100 Knots

Recommended Airspeeds (Indicated)

Minimum rate of descent	55 Knots
Maximum glide distance	80 Knots
Hydraulic failure	70 - 100 Knots
Best rate of climb	60 Knots

Altitude

(Basic Aircraft)

Maximum altitude at 5,000 lbs. (2381 Kg.)GW	20,000 Feet H_P

(FMS-28)

Maximum altitude at 5,250 lbs. (2381 Kg.) GW (with FMS-28 installed)	20,000 Feet H_P
Maximum altitude above 5,250 lbs. (2381 Kg.) GW	11,000 Feet H_D

Weights

Average empty weight (green aircraft)	2,300 Pounds
Maximum gross weight (internal) (basic aircraft)	5,000 Pounds
Maximum gross weight (internal) (with FMS-28)	5,250 Pounds
Maximum gross weight (external)	6,000 Pounds
Cargo hook rated capacity	2,650 Pounds

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Fuels

Maximum usable capacity	127.8 U.S. Gallons (490 Liters)
With auxiliary fuel tank installed	147.8 U.S. Gallons (567 Liters)
ASTM type Jet B (JP-4)	All temperatures
ASTM type Jet A or A-1 (JP-5/8)	Above –25ºF (-32ºC)
Use of 80/87 AVGAS and ASTM Jet A or A-1 mixture	No time limit
Alternate mixture 100LL AVGAS	300 hours between overhauls

Approved Lubricants

Engine gear case oils must conform to MIL-L-7808, MIL-L-23699, or DOD-L-85734 (AS) 555 series specification. Power train gear case oils must conform to MIL-L-7808, or DOD-L-85734 (AS) 555 series specification. Hydraulic fluid must conform to MIL-H-5606.

Capacity:

Engine Oil	6 U.S. Quarts (5.7 Liters)
Transmission	5 U.S. Quarts (4.7 Liters)
Tail Rotor Gear Box	33 U.S. Quarts (.31 Liters)
Hydraulic (System & Reservoir) 4 U.S. Pints	(2 Liters)

Main Rotor

Number of Blades	4
Diameter	35 Feet
Chord	10.75 Inches
Twist	-13 Degrees
Engine to Main Rotor Gear Ratio	15.23 to 1
RPM at 100%	413

Tail Rotor

Number of Blades	2
Diameter	65 Inches
Chord	6.40 Inches
Twist (Hub)	+7 Degrees
Engine to Tail Rotor Gear Ratio	2.53 to 1
RPM at 100%	2497

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GENERAL DESCRIPTION

Airframe

Overall Length	41 Feet 8.5 Inches
Overall Height	10 Feet 3.8 Inches

Passenger/Cargo Door Opening

Height	39 Inches
Width	36 Inches
Litter Door	25 Inches

Cargo Area Volume

Main Cargo Area	85 Cubic Feet
Baggage compartment	16 Cubic Feet

Terminology

WARNINGS, CAUTIONS, AND NOTES: Warnings, cautions, and notes are used throughout this manual to emphasize important and critical instructions as follows:

WARNING

An operating procedure, practice, etc., which, if not correctly followed, could result in personal injury or loss of life.

CAUTION

An operating procedure, practice, etc., which if not strictly observed, could result in damage to or destruction of equipment.

NOTE

An operating procedure, condition, etc., which is essential to highlight.

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GENERAL DESCRIPTION

Use Of Procedural Words

Concept of procedural word usage and intended meaning which has been adhered to in preparing this manual is as follows:

SHALL............................ has been used only when application of a procedure is mandatory.

SHOULD ....has been used only when application of a procedure is recommended.

MAY and NEED NOT..........have been used only when application of a procedure is optional.

WILL.........has been used only to indicate futurity, never to indicate a mandatory procedure.

Abbreviations And Acronyms

Abbreviations and acronyms used throughout this manual are defined as follows:

ADF	Automatic Direction Finder
Air Cond	Air Conditioning
A/C	Aircraft
Alt	Altitude
A/F	Airframe
ANTI COLL LT	Anti-collision light
APU	Auxiliary Power Unit
ATT	Attitude
AUX	Auxiliary
BATT	Battery
BIT	Built in Test
BL	Buttock Line
BLO	Blower
C	Celsius
CAUT	Caution
CEFA	Combined Engine Filter Assembly
CG	Center of Gravity
CKPT COMM	Cockpit Communications
CONT	Continuous
CWAP	Caution Warning Advisory Panel
DC	Direct Current
DG	Directional Gyro

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ECS	Environmental Control System
ECU	Electronic Control Unit
ELT	Emergency Locator Transmitter
ENCDG	Encoding Altimeter
ENG	Engine
F	Fahrenheit
FADEC	Full Authority Digital Electronic (Engine) Control
FS	Fuselage Station
FT	Foot, Feet
FWD	Forward
GEN	Generator
GPS	Global Positioning system
GOV	Governor
GW	Gross Weight
H_D	Density Altitude
Hg	Inches of Mercury
HMU	Hydro-Mechanical Unit
H_P	Pressure Altitude
HYD	Hydraulic
ICS	Intercommunication System
IGE	In Ground Effect
IGNTR	Igniter
IMC	Instrument Meteorological Conditions
IN	Inch(es)
IFL	Inflate
INSTR CHK	Instrument Check
INSTR LT	Instrument Light
KCAS	Knots Calibrated Airspeed
KG	Kilogram(s)
KIAS	Knots Indicated Airspeed
KTAS	Knots True Airspeed
L	Liter(s)
LB(S)	Pound(s)
LCD	Liquid Crystal Display
LDG LTS	Landing Lights
LT	Light
L/FUEL	Left Boost/Transfer Pumps
MCP	Maximum Continuous Power
MGT	Measured Gas Temperature (TOT/EGT)
MM	Millimeter(s)
MPH	Miles Per Hour (statute)
NAV	Navigation
NDOT	Rate of Ng Speed Change
NG	Gas Producer RPM (N_1)

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NP	Power Turbine RPM (N_2)
NR	Main Rotor RPM
NVM	Non Volatile Memory
OAT	Outside Air Temperature
OBS	Omni Bearing Selector
OGE	Out of Ground Effect
OVSPD	Overspeed
PAF	Pressure after Filter
PART SEP	Particle Separator
PAX	Passenger
PBF	Pressure before Filter
PF	Pressure Filter
PLA	Power Lever angle
PMA	Permanent Magnet Alternator
POS LT	Position Light
PRESS	Pressure
PRCU	Pedal Restrictor Control Unit
PSI	Pounds per Square Inch
PWR	Power
QTY	Quantity
RECP	Receptacle
RLY	Relay
RPM	Revolutions per Minute
R/FUEL	Right Boost/Transfer Pumps
SHP	Shaft horsepower
SL	Sea level
SPKR	Speaker
SYS	System
TEMP	Temperature
T/R	Tail rotor
TRANS	Transmission
TRQ	Torque
V	Volt(s), Voltage
VFR	Visual Flight Rules
VM	Volatile Memory
V_{NE}	Never Exceed Velocity
VOR	VHF Omnidirectional Range
WL	Water Line
WRN	Warning
XFR	Transfer
XMSN	Transmission
XPNDR	Transponder

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Airframe

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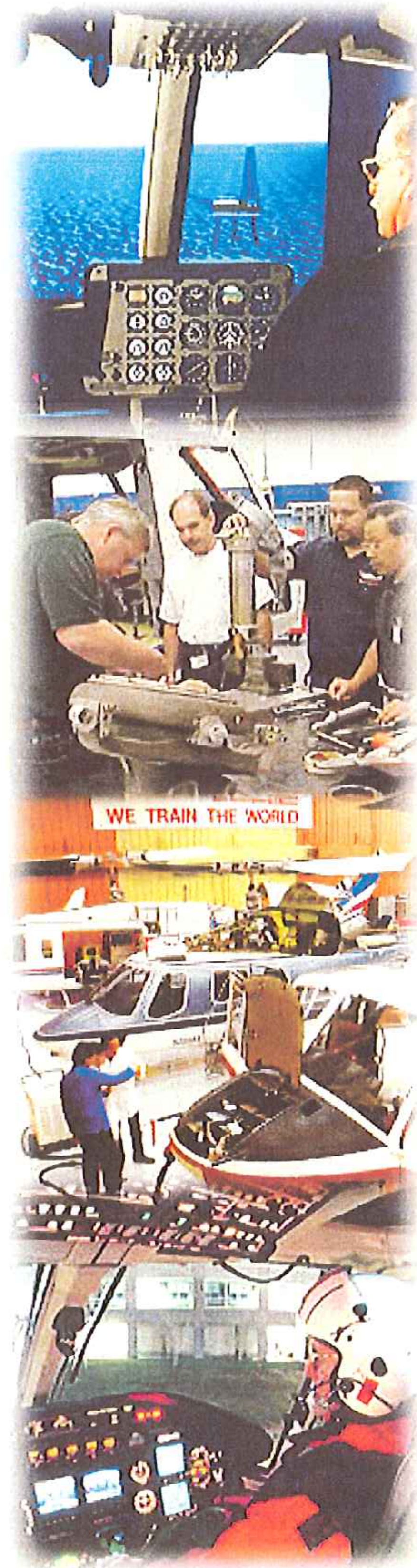

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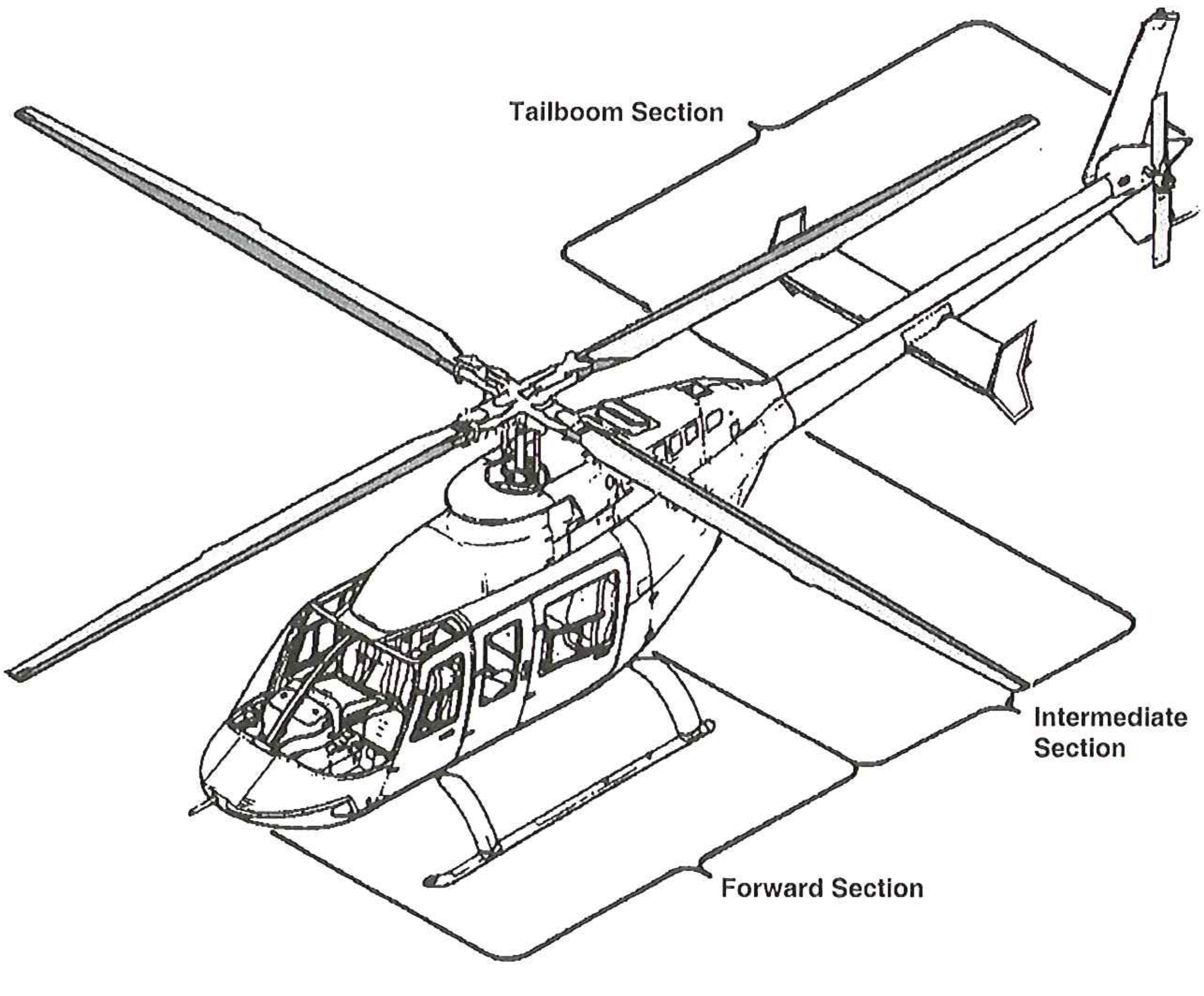
AIRFRAME

General

The fuselage consists of three main sections: the FORWARD SECTION which extends from the cabin nose to the bulkhead aft of the passenger compartment, the INTERMEDIATE SECTION which extends from the bulkhead aft of the passenger compartment to the tailboom, and the TAILBOOM SECTION.



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AIRFRAME

Forward Section

The forward section is constructed of aluminum honeycomb and carbon graphite structures and provides the major load carrying elements of the forward cabin. This section provides for pilot and passenger seating, instrumentation, electrical controls, hydraulics, fuel cell location, pylon and transmission support.

The passenger compartment is 7 inches wider (compared to 206B and L series aircraft), providing passengers with additional room. Each side of the aircraft is assembled with a tapered fairing to reduce aerodynamic drag and add additional width. The fairing begins at the forward door hinge (no additional width) and continues back to the rear bulkhead (3½ inches per side).

The 407 is equipped with five entrance/exit doors. Crew and passenger doors are located on both sides of the fuselage. Crew and passenger doors are equipped with an interior and exterior automobile style door handle. Door locks are installed in the crew, litter, passenger, and baggage compartment doors to provide security to the cockpit cabin, and baggage areas. Locks are similar in construction and are keyed alike, except for the baggage compartment door lock.

Aft of the crew doors, two passenger doors are installed on each side of the fuselage to provide access to the cabin area. Each is equipped with the same latch assembly, which may be operated from either side of the door, and a lock installed in the exterior door handle. The left passenger door is hinged on the litter door, so the two may be opened together and allow convenient access for passengers seated in the aft facing seats to enter or exit, patients on a litter (stretcher) to be loaded for ambulance transport, and large cargo to be loaded in the main cabin area. The litter door is not a structural member during flight and the aircraft may be flown without the litter door in place.

The 407 seats are covered with flame resistant fabric, leather, or other customer requested coverings. The seats are easily removable, and when removed, allow for 85 cubic feet of cargo. Designed into the forward section of the aircraft are the two fuel cells which make up the seat supports and mounting for passenger seating. A parcel shelf is located behind the forward facing passenger seats.

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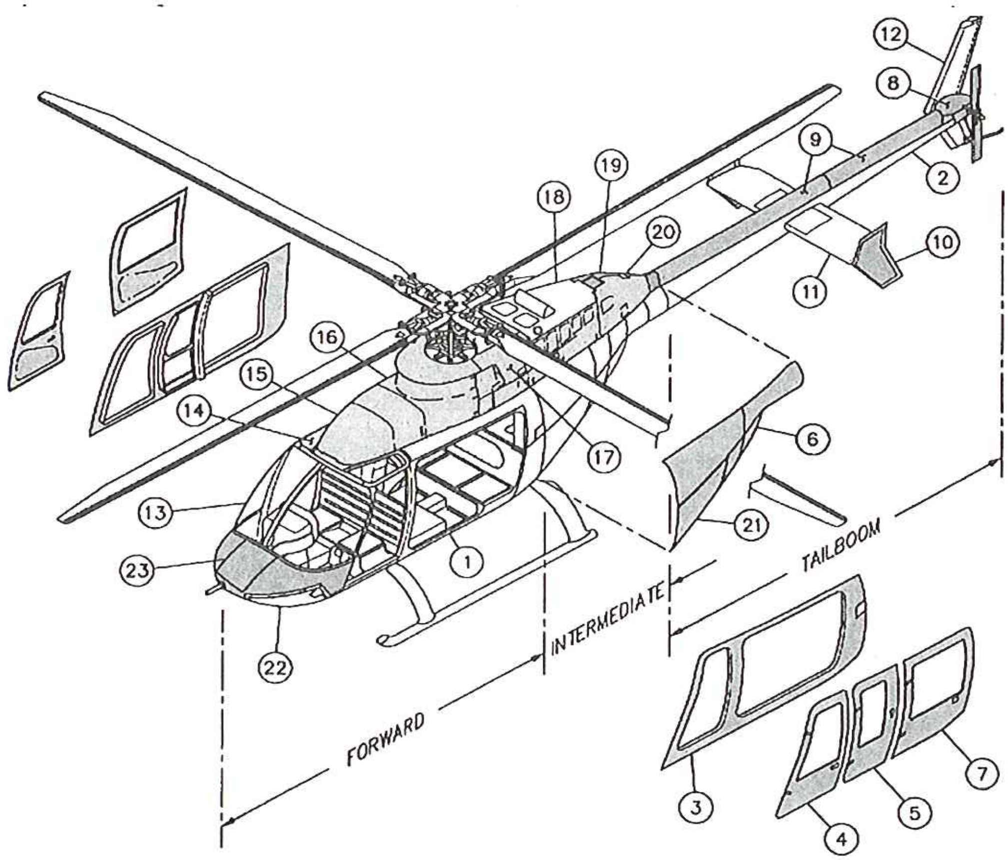
AIRFRAME

Two cabin roof windows, made of dark tinted acrylic plastic, are located over the forward crew seats and allow overhead viewing. The crew and passenger doors contain windows of light gray tinted acrylic plastic. The forward doors have a sliding window for ventilation. This window opening is adjustable by using the handle to slide the window in the window track. The handle also prevents the window from sliding out of the track.

The windshields are fabricated of tinted acrylic plastic and are supported by formed aluminum alloy sections. Two tinted transparent lower windows made from acrylic plastic are located in the lower cabin nose section.

A variety of hinged doors and panels, provide access for aircraft inspection, servicing and miscellaneous storage. The battery access door, located on the nose of the helicopter, provides access to the battery and hour meter. Two fixed landing lights are mounted below the battery and are controlled by a switch located on the pilot collective. Two of five position lights are located below the crew seats, forward of the forward landing gear cross tube. The remaining three position lights are on the tailboom. The hydraulics, transmission, landing gear, and optional cargo hook are also mounted to the forward section.



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AIRFRAME

Intermediate Section

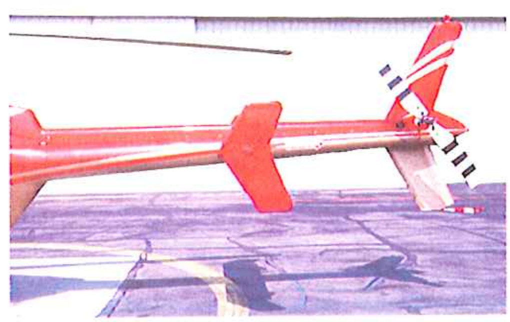
The intermediate section consists of an aluminum/composite semimonocoque construction with bulkheads for strength. This section provides a deck for engine installation, a compartment under the engine deck for heater and electrical equipment, baggage compartment located below the electrical equipment, and attachment point for the tailboom.

The engine pan and fore and aft titanium firewalls are fitted to form the engine compartment. The engine pan, located below the engine, acts as both a drip pan and firewall. The pan is curved to provide sufficient clearance to allow for the removal of accessories without removing the engine.

The equipment compartment is the area below the engine, behind the parcel shelf and above the baggage compartment. This area contains the electrical relays and regulators and has space for the optional heater or environmental control unit (a convenient location near the source of engine bleed air).

Below the engine and equipment compartment is the baggage compartment, which has a volume capacity of 16 cubic feet with a maximum floor weight of 250 pounds. An optional auxiliary (aux) fuel cell may be installed in the baggage compartment. Since the aux tank is mounted to the forward bulkhead and no fuel or fuel cell adds weight to the baggage compartment floor, all 250 pounds may be carried providing aircraft weight and balance and limitations are maintained. The baggage door, located on the left side of the fuselage is hinged at the forward end, opens the full width and height of this storage area and is secured by means of two push-button latches and a keyed lock. In-flight door security is monitored in the cockpit by a Baggage Door caution light. The compartment has ten tiedown loops to secure cargo and equipment.

Tailboom Section



As a full monocoque design, the tailboom obtains its strength through the skin and internal bracing. A doubler is bonded onto the forward right half of the tailboom, aircraft serial number 53476 and prior. Four bolts attach the tailboom to the aft fuselage. A tail rotor drive shaft, tail rotor gear box, horizontal stabilizer, two auxiliary fins, vertical fin, and three position lights are mounted to the tailboom.

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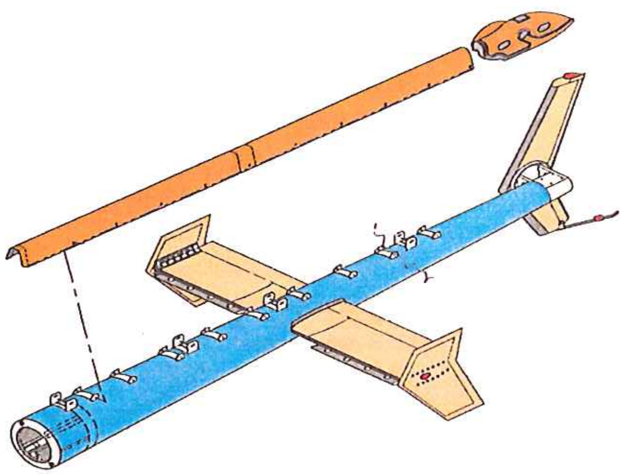
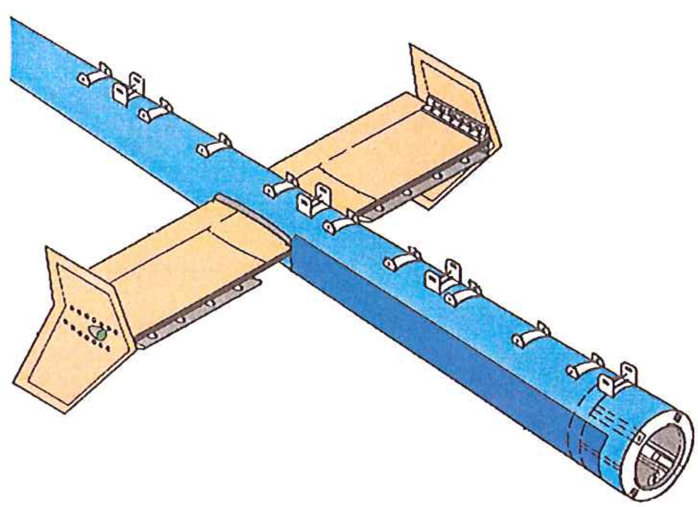
AIRFRAME

Attached to the top along the length of the tailboom are brackets that provide a mount for the hanger bearings that support the tail rotor drive shaft. Mounted to the aft end of the tailboom is the 90° tail rotor gearbox.

Mounted through the tailboom is the horizontal stabilizer. This stabilizer is a one-piece aluminum honeycomb inverted airfoil. This device provides a downward resultant lift on the tailboom to maintain the cabin in a nearly level attitude throughout all cruise airspeeds. A leading edge slat is installed to improve pitch stability during climbs. A Gurney flap on the right and left trailing edge reduces mast stress at high speeds.

Mounted to each end of the horizontal stabilizer are end plates/auxiliary fins. Each end plate is canted outboard (toe-out) 5 degrees to provide a 0 degree angle of incidence to the airflow and improve dihedral (roll) stability of the aircraft in forward flight.





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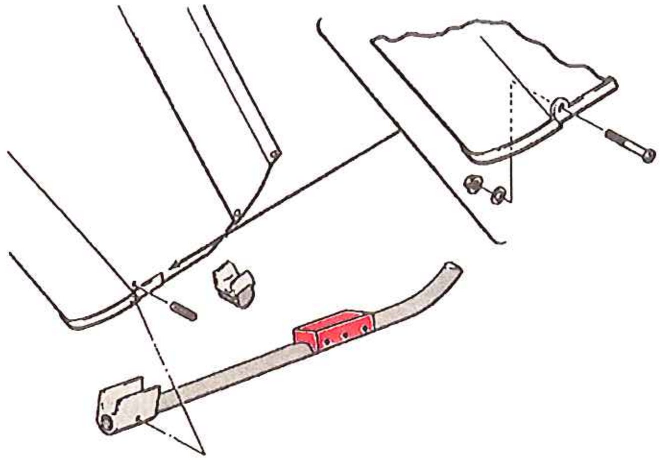
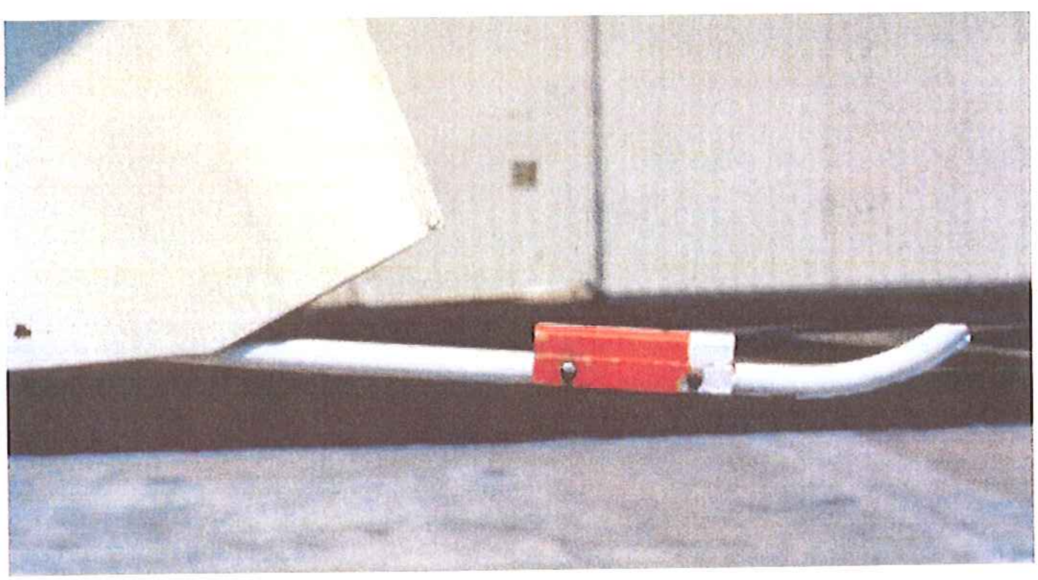


The vertical fin, composed primarily of aluminum and honeycomb construction, provides directional (yaw) stability. The leading edge is canted outboard 9 degrees to reduce the required amount of tail rotor thrust during forward flight at cruise speed. A flicker vertigo tab and anticollision light are mounted on top of the fin. A tubular tail skid is installed on the lower end. This steel tail skid and bumper are installed to warn the pilot of a tail low attitude when landing.



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AIRFRAME

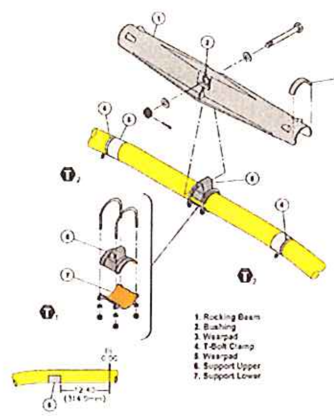
LANDING GEAR

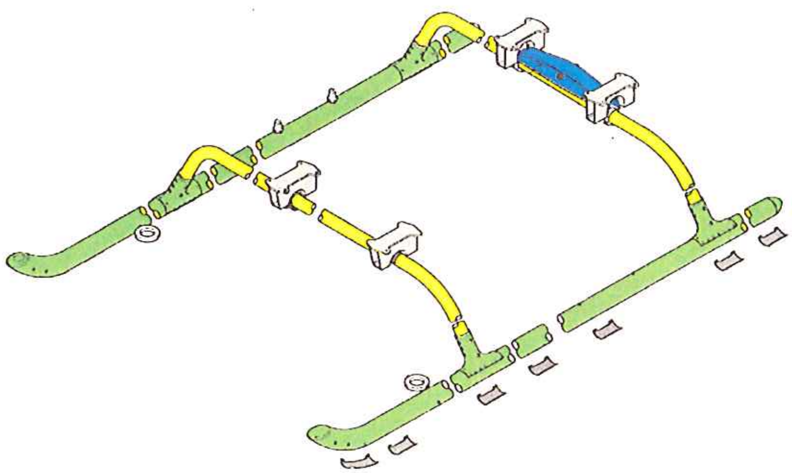
The skid type landing gear consists of two skids attached to arched cross tubes that are secured to the fuselage by means of two strap assemblies and a pivot beam. Each skid tube is fitted with a forward end step, a tow fitting, two saddles with sockets for the cross tubes, six separate shoes along the bottom, a rear cap, and two eyebolt fittings for mounting of ground handling gear.

The forward crosstube is equipped with a "WEIGHT ON GEAR" switch that is activated by flex of the forward cross tube. Through this switch three systems are energized; CYCLIC CENTERING, HOUR METER and FLIGHT TIME. When the aircraft is on the ground the cyclic centering system and related caution light are energized. The cyclic is positioned to extinguish the cyclic centering caution light. This cyclic position will minimize mast stress while the aircraft is on the ground. The hour meter and flight time are disabled when the aircraft is on the ground. When the aircraft is airborne, the cyclic centering system is disabled and the hour meter and flight time are enabled.

The aft crosstube uses a pivot beam with the attaching hardware that allows limited lateral movement of the airframe while the helicopter is on the ground. This minimizes the effect of ground resonance. During preflight the fuselage must rock freely on the beam. The landing gear can be equipped with fairings to give the cross tubes less air resistance.

It is recommended that no components be attached to the landing gear assembly except as designated by the manufacturer. Doing so could lead to failure of the crosstubes.





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AIRFRAME

Cowlings

Cowlings and fairings enclose the various roof and tailboom mounted assemblies. Cowlings provide for inspection of interior areas through the use of hinge mounts, access doors and inspection windows or cutouts. They are manufactured from composite or aluminum materials and are readily removable for maintenance access. Fairings are constructed of composite materials and reduce drag.

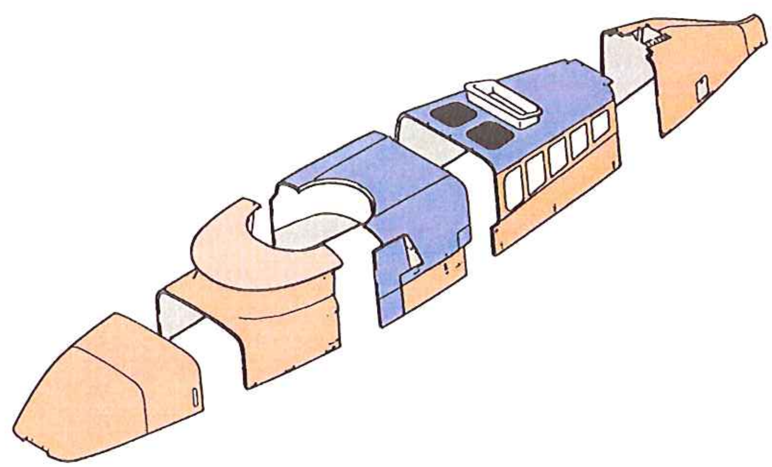
The forward cowling is composite and incorporates a hinge located at the forward end of the cowling. Support rods are attached internally on each side of the cowl and fit into roof mounted clips to support the cowl in a raised position. Two toggle hook latch fittings secure it in the closed position.

The transmission cowling is composite and encloses the forward half of the main transmission and controls. Two cutouts are provided for visual inspection of aircraft fluids. The transmission oil level cut out is on the right side and the hydraulic fluid level cut out is on the left side.

The air induction cowling is composite and encloses the aft half of the main transmission. Inlet ducts on the left and right side of the air induction cowling direct airflow into the induction screen and into the engine. Two hinged access doors, one on each side, provide for inspection of the main transmission, transmission mounting, and main drive shaft.

The engine cowling has composite hinged doors and an aluminum upper structure. The doors are held in the open position with a mechanical folding brace. The side doors and upper structure of the engine cowling incorporate screened vents to allow air movement through the engine compartment. A titanium engine exhaust stack protrudes through the upper structure. A smaller opening, facing aft and located just forward of the engine exhaust, is the engine bleed air induction port vent.

The aft cowling is composite and encloses the oil cooler, blower assembly, and engine oil tank. The cowling has a cutout to view the engine oil level. Two access doors allow viewing of the tail rotor drive shaft and engine oil servicing. Aluminum screens allow for air circulation.



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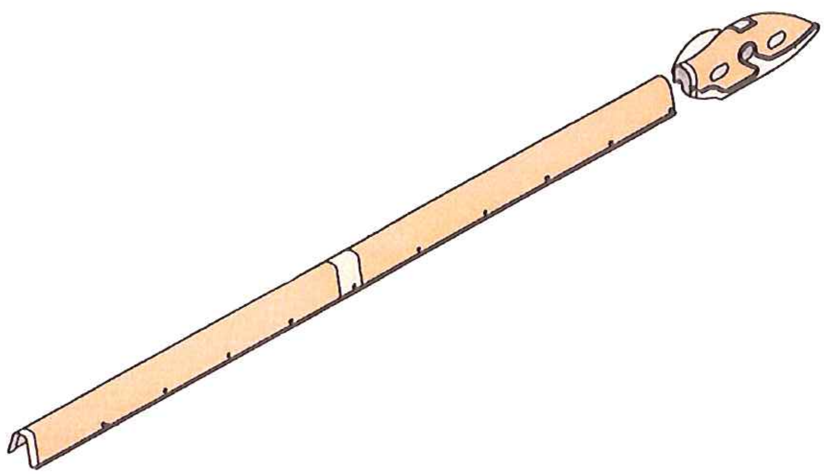
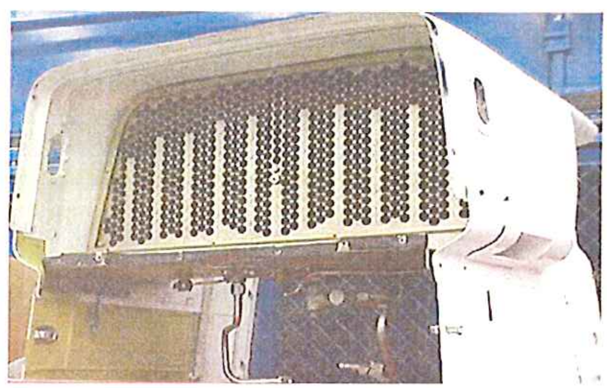
The tail rotor drive shaft is enclosed by a two piece composite cowling. The drive shaft cover is mounted in a manner that prevents the cover from making contact (chafing) with the top of the tailboom.

The composite tail rotor gearbox fairing encloses the tail rotor gearbox and is attached to the tailboom and vertical fin. It incorporates a white position light and two hinged inspection doors. The upper door is used for tail rotor gearbox oil servicing. The lower door provides access to the tail rotor gear box chip detector. A screened opening on the left side provides for gear box oil level viewing.



Particle Separator

An optional particle separator kit may be installed forward of the engine air inlet just aft of the transmission fairing instead of the induction screens. The particle separator provides continuous protection of the engine against damage from the ingestion of sand, dust, and other foreign material. The unit consists of the separator, bleed air tubing and hoses, compressor wash fittings, and other associated hardware.



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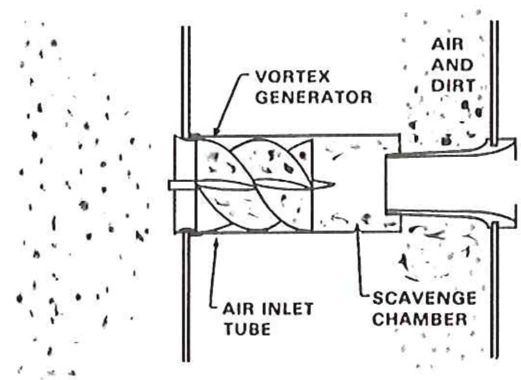

The separator section has 281 filter elements and is positioned so that all inlet air must pass through the filter elements before entering the engine. Each of the filter elements (vortex tube assemblies) in the separator consists of a vortex generator bonded into an inlet tube and a second, smaller tube to form a scavenge chamber. Materials such as hay, leaves, etc., which are too large to enter the filter elements, are stopped at the face of the separator. Particles such as dust, dirt, sand, grass, etc., enter the filter elements and are spun in the vortex generator. The particles are centrifugally routed into the scavenge chamber where they are ejected overboard through ejector tubes by engine bleed air. The separator has a tested efficiency of 85% by weight for AC coarse particles (27 micron nominal). Due to its inertial (centrifugal) action, the separator is more efficient for particles larger than 27 microns and less efficient for particles smaller than 27 microns.

Turning on the purge feature (crew compartment overhead panel switch marked (PART SEP) will take bleed air from the engine diffuser scroll and reroute this air to the base of the particle separator, through the venturi tubes, and out the ejector tubes.

A plexiglass window is installed on each side of the cowling to permit visual inspection of the separator plenum chamber between the aft side of the particle separator and the engine inlet. The ejector tubes are mounted on each side just below these windows.

The use of bleed air to purge the system will cause a slight increase in MGT indication when the particle separator is installed and turned on. When the unit is installed, approximately 6.5 pounds of weight is added to the empty weight of the aircraft.

A compressor wash fitting is installed inside the right side transmission access panel. This fitting allows aircraft operators to rinse the aircraft engine. See commercial service letter dated May 24, 2001.

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AIRFRAME

Handling and Servicing

Ground Handling

Ground handling of the helicopter consists of towing, parking, securing and mooring. Model 205 or 206 ground handling wheels are used for towing. Refer to BHT-407-MM-2 for more detailed ground handling Information.

CAUTION

DO NOT TOW THE HELICOPTER IF THE GROSS WEIGHT IS MORE THAN 5000 LBS (2270 KG) FOR MODEL 205 GROUND HANDLING WHEELS OR 4450 LBS (2020 KG) FOR MODEL 206 GROUND HANDLING WHEELS

Covers and Tiedowns

Protective covers and tiedowns are furnished as loose equipment and are used for parking and mooring of helicopter (figure 2-1). Additional equipment such as ropes, cables, devises, ramp tiedowns or dead man tiedowns will be required during mooring.

Cover — Engine Inlet

Engine inlet plug assemblies are red, flame resistant, and each cover is attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To install the engine Inlet plug, make sure that the side marked TOP is up. Push the engine Inlet plug Into the engine air inlet.

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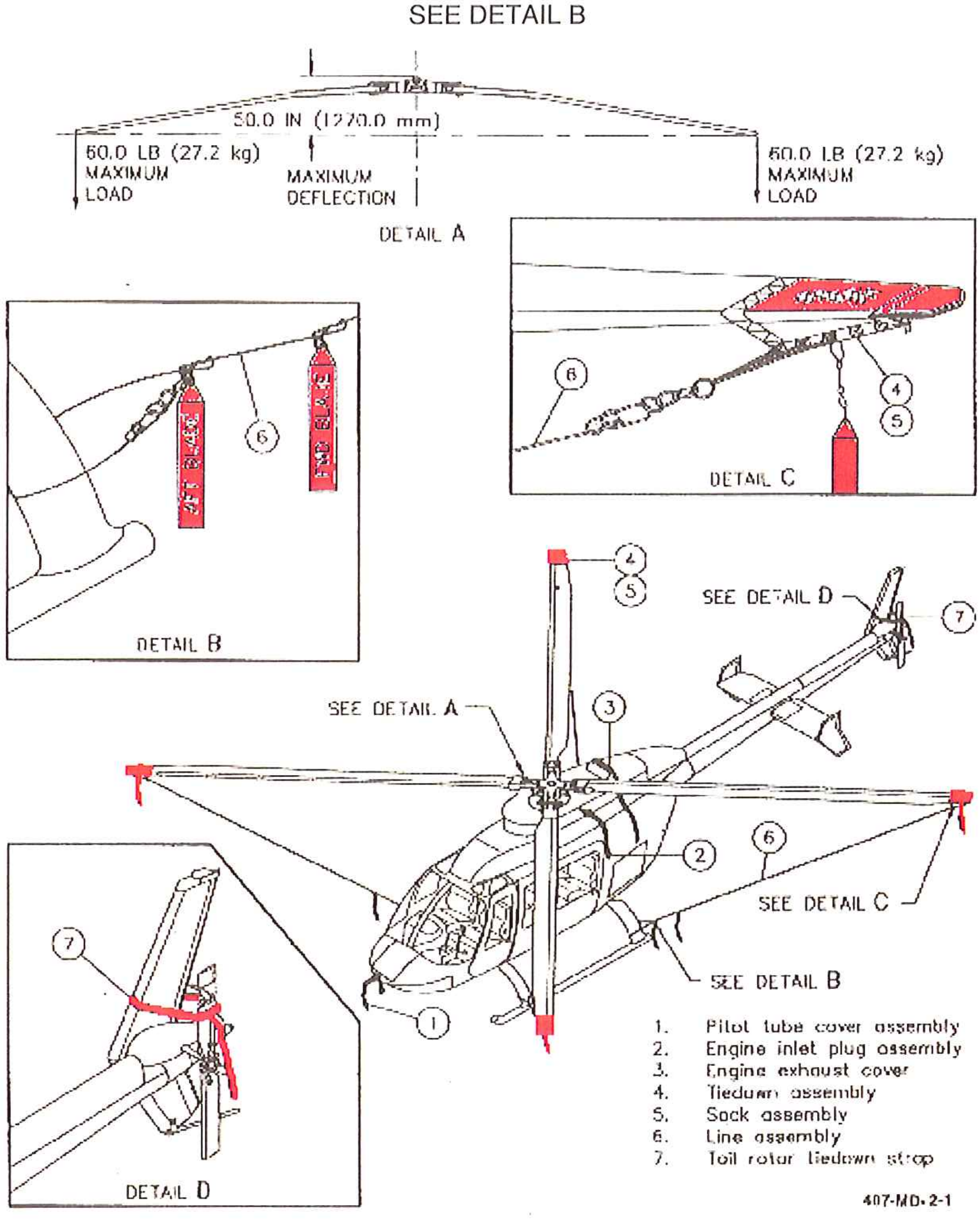

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Cover — Pitot Tube

Pitot tube cover assembly is red, flame resistant and attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To Install pitot tube cover, push it over the pitot tube. Attach tie cord.

WARNING

THE PITOT TUBE CAN BE HOT.

Cover — Engine Exhaust

Engine exhaust cover is red, flame resistant, and includes a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. A 1/4-inch diameter elastic tie cord is attached to cover for securing to engine exhaust. To install the engine exhaust cover, push it over the exhaust tailpipe. Attach the cord.

Tiedown — Main Rotor

For each main rotor blade, there is a main rotor tiedown assembly. Each tiedown assembly has a sock assembly and a line assembly. Use these to attach the blades to the landing-gear cross tubes. The sock assembly is red and has a red streamer, stenciled with white letters REMOVE BEFORE FLIGHT, attached to it. The line assembly is made of 0.19 inch (4.83 mm) diameter nylon and has a ring and an attached flag. The flag is stenciled with the letters FWD BLADES or AFT BLADES.

Install the main rotor tiedown assemblies as follows:

CAUTION

DO NOT CAUSE THE MAIN ROTOR BLADES TO BEND MORE THAN THE LIMITS SHOWN IN FIGURE 2-1 DETAIL A.

At the same time you align the main rotor blades, align the tail rotor blades with the vertical fin. This will make it possible to install the tail rotor tiedown.

1. Turn the main rotor blades until there are two blades aft of the fuselage station of the main rotor hub. When you look down on the helicopter, the four blades make an X over the vertical center line of the fuselage.

2. Install the two FWD BLADES sock assemblies on the ends of the main rotor blades that are forward of the fuselage station of the main rotor hub.

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3. Put a line assembly around each outboard end of the forward cross tube of the landing gear.

NOTE

Rings are pre-set to apply the necessary tension to the forward and aft main rotor blades.

4. Attach the snaps of the two line assemblies to the rings of the two FWD BLADES sock assemblies.

5. Install the two AFT BLADES sock assemblies on the ends of the two aft main rotor blades.

6. Put a line assembly around each outboard end of the cross tube of the landing gear.

7. Attach the snaps of the two line assemblies to the rings of the two AFT BLADES sock assemblies.

TIEDOWN — TAIL ROTOR

The tail rotor tiedown strap is made of 0.025 x 1.0 x 92.0 inch (0.6.35 x 25 x 2340 mm) nylon webbing. It is red and stenciled with white letters REMOVE BEFORE FLIGHT.

Install the tail rotor tiedown as follows:

CAUTION

DO NOT TIE DOWN TAIL ROTOR TO EXTENT THAT TAIL ROTOR BLADE FLEXES.

1. Turn the main rotor blades until there are two blades aft of the fuselage station of the main rotor hub. When you look down on the helicopter, the four blades should make an X over the vertical center line of the fuselage. Align the tail rotor with the vertical fin.

2. Install the tail rotor tiedown strap in the loop on the lower left side of the vertical fin.

3. Wind the tail rotor tiedown strap around the tail rotor blade.

4. Attach the tail rotor tiedown strap to the loop on the lower left side of the vertical fin.

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Bell Helicopter

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Crew Compartment

Training Academy

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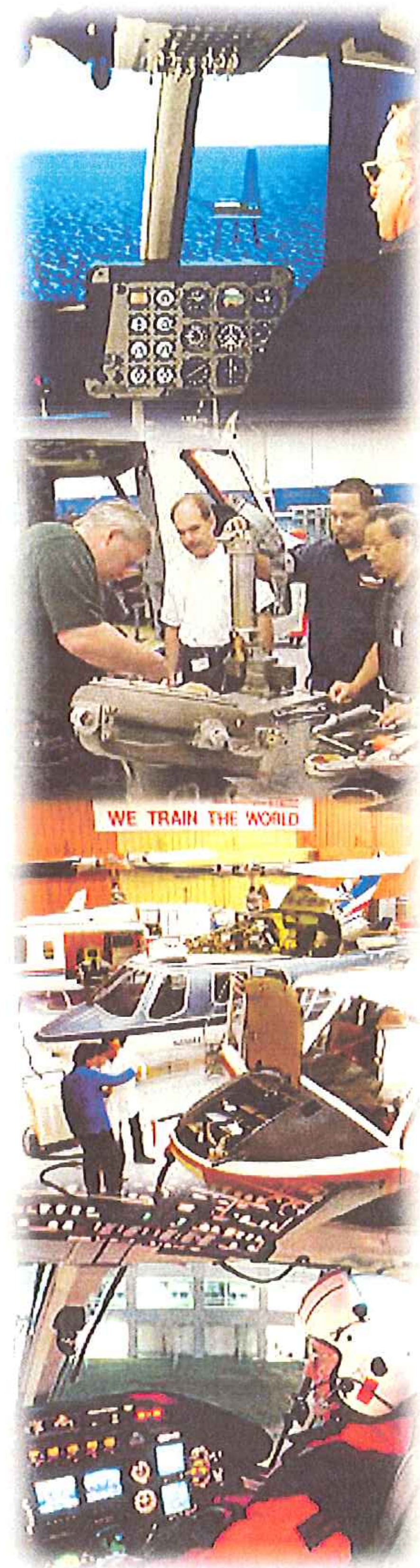

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CREW COMPARTMENT

General

The pilot's station is situated on the right side of the crew compartment. The left side can accommodate a passenger, or a copilot if dual controls are installed.

The instrument panel is mounted on a center pedestal forward of the two crew seats and is tilted forward at a five degree angle for maximum visibility. Flight instruments are located on the right side of the panel and system instruments are in two rows to the left of the flight instruments. Caution, Warning, and Advisory Panel (CWAP) lights are mounted just below the glare shield across the top of the instrument panel.

The pedestal extends aft from the instrument panel between the seats to form a console for the radios.

The overhead console is centrally positioned aft of the windshields on the cockpit ceiling. This console contains most of the circuit breakers and electrical switches.

Instrument system

The instrument system is divided into four separate categories: flight, navigation, propulsion, and miscellaneous. All indicators are installed in the hinged instrument panel except the standby compass and hourmeter. The standby compass is mounted to the right side of the cabin structure slightly forward of the instrument panel. The hourmeter is typically mounted in the nose compartment.

The flight instruments include the pitot/static system, airspeed indicator, altimeter, turn and slip indicator, and vertical speed indicator.

The navigation instrument on the basic 407 helicopter is the standby magnetic compass.

The propulsion instruments are the dual tachometer, gas producer tachometer, engine torquemeter indicator, measured gas temperature indicator, engine oil pressure/temperature indicator, transmission oil pressure/temperature indicator, and fuel quantity indicator.

The miscellaneous instruments are the clock/OAT/voltmeter and the engine hourmeter.

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CREW COMPARTMENT

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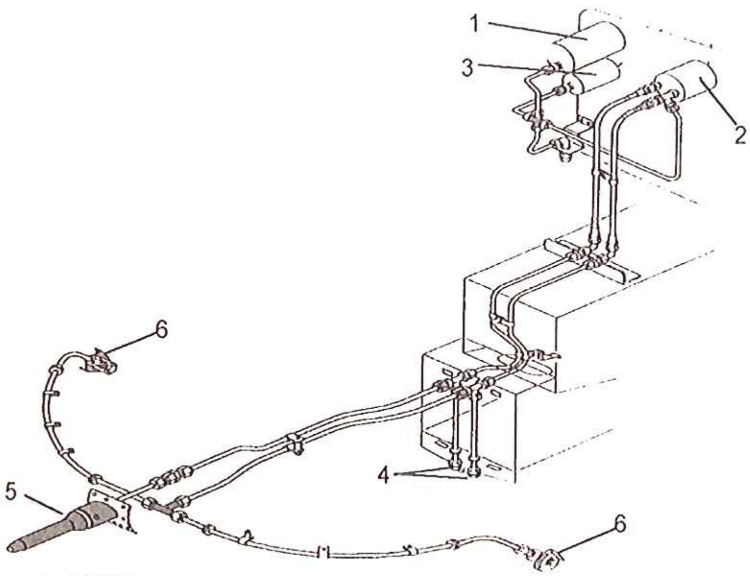
CREW COMPARTMENT

Pitot /Static System



1. Altimeter
2. Airspeed
3. Vertical speed indicator
4. Pitot static drains
5. Pitot tube (Heated)
6. Static ports (Heated)

The pitot tube is mounted on the most forward part of the cabin nose structure. The tube supplies impact air to the airspeed indicator. Static air pressure for instrument operation is obtained from two heated static ports on the left and right sides of the aircraft, aft of the cabin lower windows. Turning the PITOT HEAT switch to the ON position activates the heater elements for both the pitot tube and static ports. Internal and drain lines for the pitot tube and static ports are made of aluminum.

An airspeed activated pedal stop system uses pitot air pressure to electronically control a pedal stop solenoid. This solenoid will extend the pedal stop restrictor into the left pedal range of travel. This restrictor will allow full left pedal at less than 50 knots IAS. When airspeed increases to 55 knots IAS and above, left pedal travel is limited to less than full travel. A status press-to-test switch / annunciator and a PEDAL STOP caution light have been added to indicate normal or abnormal status.

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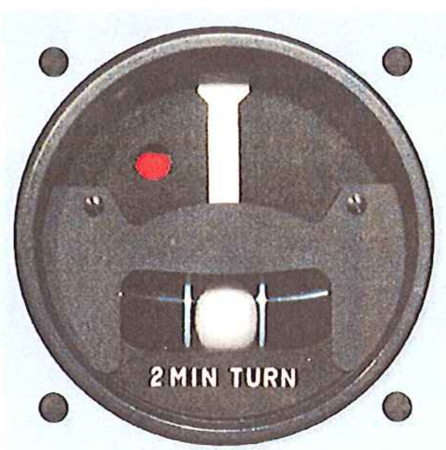
Airspeed Indicator - The airspeed indicator is a standard pitot/static instrument. This indicator provides an airspeed reading in knots (KTS) by measuring the difference between impact air pressure from the pitot tube and the static air pressure from the static ports.

The indicator presents airspeed from 0 to 160 knots and is scaled in 5 knot increments at 20 knots and above. A maximum autorotation speed red / white line is located at 100 knots and a maximum speed (Vne) red line is located at 140 knots.

Altimeter - The barometric pressure altimeter presents an altitude reading in feet above mean sea level (MSL) based on the relationship between the static air pressure and the barometric setting on the altimeter. The barometric setting may be adjusted to reflect the current barometric pressure corrected to sea level in inches of mercury or in millibars, depending on the instrument installed.



Turn And Slip Indicator - The inclinometer is a simple instrument consisting of a curved glass tube, ball, and dampening fluid. The ball indicates when the helicopter is in a coordinated turn or balanced straight and level flight, i.e. "in trim". If the helicopter is in a slip or skid, the ball will move off center. The turn needle is powered by the 28 Vdc bus through the TURN circuit breaker switch. The turn needle indicates the rate at which the aircraft is turning about the vertical axis degrees per second. A red electrical power failed flag indicates the turn needle is inoperative.

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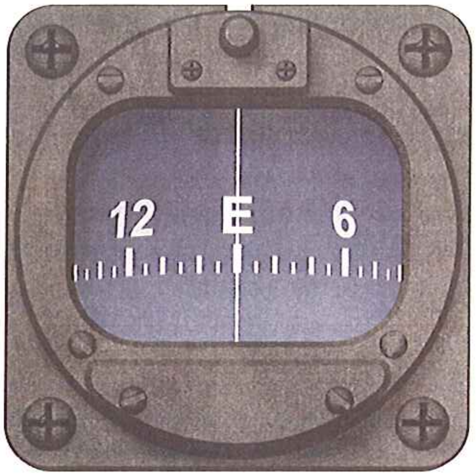
Navigation Instruments



Standby Magnetic Compass - The magnetic compass is a standard, non-stabilized, magnetic type instrument mounted on a support attached to the pilot side of the forward cabin next to the door post. A compass correction card is located below the instrument.

Propulsion Instruments

Dual Tachometer (Np / Nr) - The Dual Tachometer is equipped with two pointers, one displaying Main Rotor RPM on the outer scale and one that displays engine RPM on the inner scale. All scales are in percent of rated RPM.

The Np (engine) side of the instrument is powered by the 28 Vdc bus through its own circuit breaker.

An Np monopole speed pickup mounted on the engine provides two separate signals to the ECU. One signal is the primary driver of the Np gage. A second signal is the primary input to the ECU.

The Nr (rotor) side of the gage is powered by the 28Vdc bus through its own circuit breaker. An Nr monopole speed pickup mounted on the transmission lower case provides three separate identical signal outputs of rotor Rpm. One signal is sent to the Nr circuit of the dual tachometer gage. One signal is sent to the ECU and the other is sent to the main rotor RPM sensor switch.

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Gas Producer Tachometer (Ng) - The gas producer (Ng) displays engine gas producer speed in per cent of rated RPM and is powered by the 28 Vdc bus through its own circuit breaker. An Ng monopole speed pickup mounted on the engine accessory gearbox provides two separate signals to the ECU. One signal is for the ECU, the other is for the gage.

Ng in percent RPM is displayed on the LCD trend arcs and the digital display. The gage has the ability to record exceedances.

Engine Torque Meter (TRQ) Indicator - The engine torque meter utilizes engine oil pressure to determine power output of the engine in the form of percent torque. The gage is powered by the 28Vdc bus through its own circuit breaker. Oil pressure is fed to both the FADEC and airframe engine oil pressure transducers. The airframe transducer provides a signal to the torque gage. TRQ in percent is displayed on the LCD trend arcs and digital display. The torque gage has the ability to record exceedances. The top of the green arc (93.5%) on the torque gage represents 630 shaft horsepower at 100% Np. The red line (100%) on the torque gage represents 674 shaft horse power at 100% Np.

Measured Gas Temperature (MGT) Indicator - The measured gas temperature (MGT) indicator displays engine gas temperature in degrees Celsius. Four probes measure gas temperature between the gas producer turbine and the power turbine. MGT is displayed on the LCD trend arcs and on the digital display. The MGT gage is powered by the 28 Vdc bus through its own circuit breaker. The MGT gage can determine and record two separate exceedance levels, one for starting and another for normal operations.

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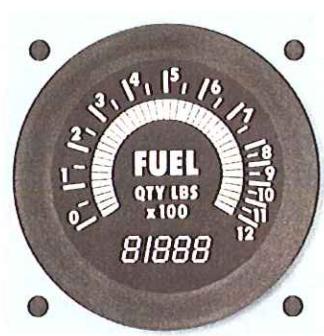
Engine Oil Pressure/Temperature Indicator - The engine oil pressure/temperature indicator displays oil pressure in PSI on the left side of the instrument and oil temperature in degrees Celsius on the right side. This information is displayed on the LCD trend arcs.

The 28 Vdc bus, through its own circuit breaker, powers each side of the indicator

The engine oil temperature input signal is provided by a thermo-bulb installed on the engine oil tank. A transducer, mounted on the forward right side of the engine firewall, provides engine oil pressure.

Transmission Oil Pressure/Temperature Indicator - The transmission oil pressure/temperature indicator displays oil pressure in PSI on the left side of the instrument and oil temperature in degrees Celsius on the right side. This information is displayed on the LCD trend arcs. The 28 Vdc bus, through its own circuit breaker, powers each side of the indicator.

The transmission oil temperature input signal is provided by a thermo-bulb installed on the transmission oil filter manifold. A transducer, mounted on the transmission deck oil manifold, provides the transmission oil pressure input signal.



Fuel Quantity Indicator - The fuel quantity indicator displays total fuel quantity in pounds from the three quantity probes located in the fuel cells. The fuel quantity is displayed on the LCD arcs and on the digital display. The fuel quantity indicator is powered by the 28 Vdc bus through its own circuit breaker.

The fuel quantity gage indicates the total usable fuel in both fuel tanks. If an auxiliary tank is installed, the auxiliary quantity is automatically computed into the aft tank quantity.

When the FUEL QTY FWD TANK button is pushed, the fuel gage will indicate the fuel quantity in only the forward tank.

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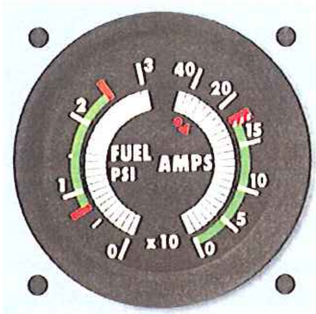
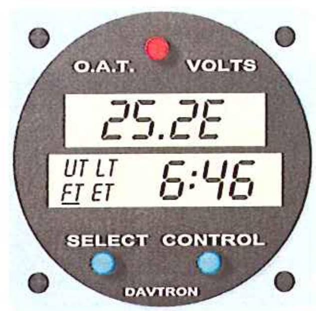
CREW COMPARTMENT

Miscellaneous Instruments



Fuel Pressure/Ammeter – The fuel pressure/DC ammeter indicator displays fuel pressure in PSI on the left side of the instrument and DC current load on the generator on the right side of the instrument, on the LCD trend arcs. Fuel pressure is derived from a pressure transducer located below the fuel valve and measures the output of both boost pumps. The ammeter indicates the load in amperes that is being supplied to the 28Vdc bus by the generator. Each side of the indicator is powered by the 28 Vdc bus through its own circuit breaker.

Clock/Oat/Voltmeter - The clock/OAT/Voltmeter is a multifunction indicator mounted in the upper left area of the instrument panel. The clock is powered by the 28 Vdc bus through its own circuit breaker. When the battery switch is turned on, the instrument defaults to the VOLTS mode and displays voltage available at the bus. A red button located on the top center of the instrument selects outside air temperature (OAT) in either degrees Celsius or Fahrenheit and VOLTS. The lower half of the instrument is a chronometer and displays:



Universal Coordinated (ZULU) Time (UT) in 24 hour format.
Local Time (LT) in 12 or 24 format
Elapsed Time (ET) count up timer to a maximum of 99 hours, 59 minutes.
Elapsed time countdown from 59 hours, 59 minutes.
Flight Time (FT) count up or down a maximum of 99 hours, 59 minutes. The flight time feature is activated when the aircraft is in flight, i.e. weight off gear.

Engine Hourmeter - The engine hourmeter is typically mounted on the aft bulkhead of the battery compartment. The hourmeter is powered by the 28 Vdc bus through its own circuit breaker. The system is activated when the Ng is greater than 55% and the aircraft is in flight, i.e. weight off gear.

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Exceedance Monitoring

Exceedances are defined as limits of operation above which maintenance action may be required.

TRQ, MGT, and Ng gages contain a microprocessor with a 10 year lithium backup battery that has the ability to record in its Non-Volatile Memory (NVM) up to 50 events or exceedances. For each event, the gage records the date, duration, and peak value of the exceedance.

The CHECK INSTR caution light will illuminate and the gage LCD trend arcs will begin flashing to provide the pilot with an advance warning that an exceedance is about to be recorded. If the pilot makes control inputs to reduce the instrument readings below the advisory values, the gage trend arcs return to normal (i.e. stop flashing), and the CHECK INSTR will extinguish. No exceedance will be recorded.

If a limit is exceeded, the gage trend arcs will stop flashing and an "E", plus the exceedance value, will appear in the digital readout. The CHECK INSTR caution light will remain on until the pilot acknowledges the exceedance by pushing the INSTR CHK button. Once the pilot pushes the INSTR CHK button, the numeric value of the last recorded exceedance will appear for up to 11 seconds on the gage digital readout. Releasing the INSTR CHK button causes the "E" and the exceedance value to disappear from the digital display and the CHECK INSTR light will extinguish. The instrument will then function normally. The "E" will only re-appear after the BATT switch is turned OFF and then back to BATT (ON).

An "E" and the value will continue to display each time the gage is powered up until the exceedance(s) is/are removed from the NVM of the gage. This information will be stored in the instruments NVM until cleared with a computer.

If an exceedance has been recorded by the gages, the pilot shall review the aircraft maintenance logbook to verify that an entry regarding the exceedance has been entered in the logbook and that corrective maintenance has been performed prior to operating the helicopter. If corrective action has not been taken, the aircraft is considered not airworthy. Refer to the aircraft maintenance manual and maintenance personnel for corrective action. Corrective maintenance action shall be performed to return the aircraft to service. If the affected gage has not had the exceedance cleared with the LITTON furnished software, the "E" will continue to appear each time the aircraft is powered up. The pilot can verify the exceedance value by pressing the INSTR CHK button. The exceedance value will disappear from the display when the INSTR CHK button is released. The instrument will then function normally. The "E" will only re-appear after power has been removed from and reapplied to the gage.

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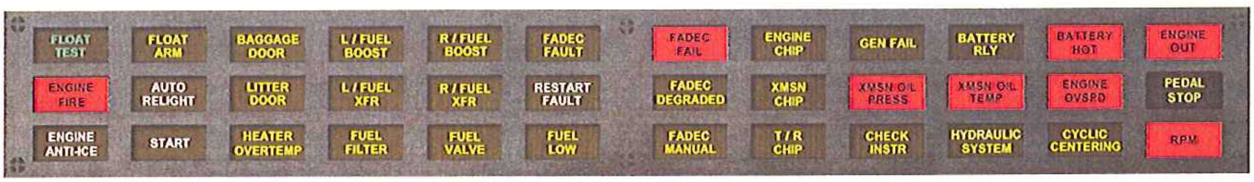
CREW COMPARTMENT

Caution And Warning System

The caution and warning system consists of the Caution Warning and Advisory Panel (CWAP), engine RPM sensor and warning horn, low rotor RPM sensor and warning horn, and the FADEC Warning horn.

Caution Warning and Advisory Panel (CWAP)

The CWAP is powered by the 28Vdc bus and protected by its own circuit breaker. Each segmented indicator light is in series with its respective system. Warning lights are colored RED, Caution lights are colored AMBER, and Advisory lights are colored WHITE or GREEN. Pressing the C/W LT TEST button will illuminate the CWAP and the FADEC MODE SWITCH.



Definitions

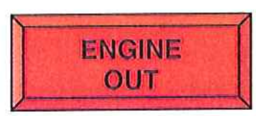
Land As Soon As Possible

Land without delay at the nearest suitable area (i.e. open field) at which a safe approach and landing is reasonably assured.

Land As Soon As Practical

The landing site and duration of flight are at the discretion of the pilot. Extended flight beyond the nearest approved landing area is not recommended.

Engine Out



The ENGINE OUT warning system provides both visual and audible indications of an engine out condition. This includes an ENGINE OUT warning light, located on the CWAP, and the ENGINE OUT warning alarm (intermittent sound), located on the overhead console.

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The FADEC Electronic Control Unit (ECU) controls the engine out relay in AUTO mode. When gas producer RPM drops below 55 ± 1%, the ECU will close the ENGINE OUT relay, activating the warning light and alarm. In the FADEC manual mode, the ENGINE OUT relay is connected directly to the Ng speed sensor. When the gas producer RPM drops below 55 ± 1% the engine out relay closes, activating the warning light and alarm.

RPM

The low rotor RPM warning system provides both visual and audible indications of a low and high rotor RPM condition. The low rotor RPM light and alarm (continuous tone) are activated when the Nr sensor detects the rotor RPM decelerating below 95%.

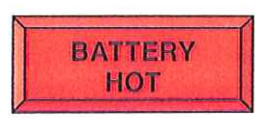
The rotor RPM warning light will illuminate when rotor RPM exceeds 107%. The RPM warning audio will not activate for high RPM.

Warning Audio Mute System

A button is located in the upper right hand corner of the instrument panel to silence the audio alarms for ENGINE OUT and LOW ROTOR RPM. Pressing the HORN MUTE button will temporarily silence both warning tones. The warning horns are automatically reset to operate when preset operational parameters are reestablished.

Pedal Stop

The PEDAL STOP amber caution light will illuminate when the electro-magnetically operated pedal stop mechanism fails to engage or disengage. The PEDAL STOP caution light will illuminate if the pedal stop fails to engage when the airspeed accelerates above 55 KIAS or if the pedal stop fails to disengage when airspeed decelerates below 50 KIAS. This caution light is part of the airspeed activated pedal stop kit installation. A hand operated mechanical release is provided to manually disengage the pedal stop mechanism.



Battery Hot

A temperature probe, installed in the aircraft battery, activates the BATTERY HOT warning light when preset temperature limits (approximately 63°C for the 17 amp/hour battery and 71°C for the 28 amp/hour battery) are reached.

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Battery Relay

The BATTERY RELAY caution light illuminates when the battery switch is placed in the OFF position and the battery relay remains closed.

Engine Overspeed

The ENGINE OVSPD warning light indicates an engine overspeed condition. The light is activated when Ng exceeds 110% or Np versus TQ is above the maximum continuous limit. (102.4% Np at 100% TQ to 108.6% Np at 0% TQ).

Cyclic Centering

The CYCLIC CENTERING caution light illuminates when a sensor connected to the pilot's cyclic detects that the cyclic is not centered when the aircraft is on the ground. To minimize mast stress at the main rotor hub, the pilot should keep the cyclic centered when the aircraft is on the ground. The cyclic centering system is enabled through the WEIGHT ON GEAR switch and is not functional when the aircraft is in flight.

Hydraulic System

The HYDRAULIC SYSTEM caution light illuminates when hydraulic pressure decreases below 650 PSI. The light extinguishes when the pressure increases above 750 PSI.

Generator Failure

The GEN FAIL caution light illuminates when the generator is not on line or has failed.

Check Instrument

The CHECK INSTR caution light will illuminate when TQ, Ng, or MGT has had, or is about to have, an exceedance.

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Transmission Oil Pressure

The transmission oil pressure warning system includes the TRANS OIL PRESS warning light, a pressure transducer, and an oil pressure switch that are mounted in the transmission oil manifold assembly. The light will illuminate when pressure decreases to 30 PSI ±2 and extinguishes when pressures increases to 38 PSI.

Transmission Oil Temperature

A thermal switch is located on the transmission-mounted oil manifold. If the oil temperature rises above 110°C ± 5°C, the XMSN OIL TEMP warning light will illuminate.

Engine Chip Detector

The engine chip detector caution system includes the ENGINE CHIP caution light, and two magnetic chip detectors. If ferrous metal particles are found in the engine oil, the magnets in the detectors will attract them, completing the circuit and illuminating the ENGINE CHIP caution light.

Transmission Chip Detector

The transmission chip detector caution system includes the XMSN CHIP caution light, two transmission chip detectors and a freewheeling unit chip detector. If ferrous metal particles are found in the transmission oil, the magnets in the detectors will attract them, completing the circuit and illuminating the XMSN CHIP caution light.

Tail Rotor Gearbox Chip Detector

The tail rotor gearbox chip detector caution system includes the T/R CHIP caution light and the tail rotor gearbox chip detector. If ferrous metal particles are found in the gearbox oil, the magnet in the detector will attract them, completing the circuit and illuminating the T/R CHIP caution light.

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FADEC Fail

The FADEC FAIL warning light indicates a failure of the FADEC system during start or in normal engine operation. A FADEC FAIL warning light will illuminate and the FADEC FAIL horn will sound when the ECU has detected a failure of the FADEC system.

FADEC Degraded

The FADEC DEGRADED caution light receives its signal from the ECU and indicates a degraded condition. The FADEC DEGRADED caution light may also illuminate during engine shutdown, indicating that a FADEC related error code has been recorded by the ECU.

FADEC Manual

The FADEC MANUAL caution light receives its signal from the ECU when the FADEC is operating in manual mode. The AUTO RELIGHT white advisory light will also be illuminated.

FADEC Fault

The FADEC FAULT caution light receives its signal from the ECU. This light indicates that MGT, Np, or Ng automatic limiting circuits may not be functional or that the permanent magnet alternator (PMA) has failed.

Restart Fault

The RESTART FAULT light is a white advisory light. When it illuminates, a subsequent automatic engine start may not be possible.

Fuel Low

The FUEL LOW caution light illuminates when approximately 110 ± 15 pounds of fuel remains in the aft (main) cell. A fuel low sensor located in the aft fuel cell, independent of the fuel quantity indicating system, controls the FUEL LOW caution light.

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Fuel Boost Pumps

THE L/FUEL BOOST and R/FUEL BOOST caution lights are controlled by their respective fuel pressure switch located at the base of each pump. The switches are activated when fuel pump pressure drops to 1.5 PSI ± 0.5 or lower, indicating a pump failure. The lights extinguish when fuel pressure increases to 5 PSI and above.

Fuel Transfer Pumps

THE L/FUEL XFR and R/FUEL XFR caution lights are controlled by their respective fuel pressure switch located at the base of each pump. The switches are activated when fuel pump pressure drops to 1.5 PSI ± 0.5 or lower, indicating a pump failure. The lights extinguish when fuel pressure increases to 5 PSI and above. In the case of dual transfer pump failure, up to 135 pounds of Jet B fuel or 151 pounds of Jet A fuel may be trapped (unusable) in the forward tank.

Aircraft SN 53001 to 53174 have transfer caution lights that illuminate when fuel transfer from the forward tank is nearing completion. After transfer is complete, both caution lights will extinguish. In aircraft SN 53175 and subsequent, the caution lights illuminate only in the event of a pressure drop indicating a pump failure. There is no indication that fuel transfer is completed.

Fuel Valve

The FUEL VALVE caution light illuminates when the fuel valve position differs from the switch indication or when the valve is in transit between the open and closed positions.

Fuel Filter

The FUEL FILTER caution light illuminates when the airframe fuel filter is in an impending bypass condition (approximately 1 PSI differential). The airframe fuel filter will bypass at approximately 4 PSI differential.

Baggage Door

A microswitch on the baggage door illuminates the BAGGAGE DOOR caution light when the door is opened or is not securely fastened.

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Litter Door

The litter door caution system includes the LITTER DOOR caution light, microswitches for the upper and lower door strikers, and related wiring. If the litter door is not securely fastened or if it has been removed, the caution light illuminates.

Heater Overtemp

The HEATER OVERTEMP caution light receives its signal from one of three temperature sensors, one each below the pilot and copilot seats and one located in the passenger compartment heater ducting. The HEATER OVERTEMP caution light will illuminate when the temperature in any of the sensors reaches 104° C.

Auto Relight

The AUTO RELIGHT advisory light illuminates whenever the engine igniter is operating. During the start sequence, the AUTO RELIGHT advisory remains on until Ng reaches 60% or when the FADEC is operating in the manual mode. The AUTO RELIGHT will illuminate in flight if the ECU detects an Ng deceleration. When the start switch is engaged, the AUTO RELIGHT and the START white advisory lights confirm a FADEC automatic start.

Start

The START light is a white advisory light that illuminates when the start relay is energized. The START light will extinguish when Ng reaches 50%.

Engine Anti-Ice

The ENGINE ANTI-ICE white advisory light will illuminate when the engine anti-icing system is operating. This light is standard on ship serial numbers 53095 and subsequent.

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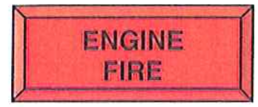


Engine Fire (FMS-21)

The ENGINE FIRE warning light and the fire test button are installed with the optional engine fire detector kit. The ENGINE FIRE caution light will illuminate when a heat sensitive wire mounted on the inboard side of the engine compartment senses excessive heat. The ENGINE FIRE warning light will illuminate when the press-to-test FIRE DET TEST button is pressed.

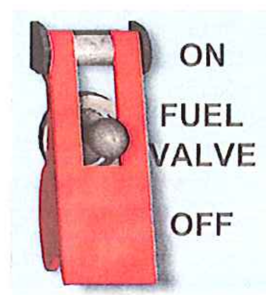
Float Arm (FMS-1)

The FLOAT ARM caution light illuminates when the pilot selects the "protected" float arm switch located on the collective head. The floats provide the aircraft with a means of landing the aircraft on water should an emergency occur during flight. This float kit has an airspeed restriction of 60 KIAS or less for inflation. Refer to FMS-1 for additional limitations and emergency procedures.

Float Test (FMS-1)

The FLOAT TEST advisory light will illuminate when the float test switch and the float inflation switch are simultaneously pressed and held.

Fuel Valve



A fuel valve switch is located on the lower right hand side of the instrument panel and electrically operates a motor driven fuel valve providing a means of shutting off fuel to the engine. When the FUEL VALVE switch is moved to the ON position, a transition advisory caution light (FUEL VALVE) will momentarily illuminate to advise the pilot the fuel valve is moving to the open position. Once the valve reaches the open position, the caution light will extinguish. When the FUEL VALVE switch is moved to the OFF position, the caution light will momentarily illuminate to advise the pilot the fuel valve is moving to the closed position.

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Electrical Fail Safe Systems

Engine Anti-Ice System - The hot air solenoid valve is powered from the 28 Vdc bus through the ENGINE ANTI-ICE circuit breaker and the ENG ANTI-ICE switch. When the solenoid valve is de-energized (ENG ANTI-ICE switch is ON), bleed (hot) air passes from the compressor diffuser through the anti-ice valve to the engine inlet housing. This hot air aids in preventing ice formation on the hollow inlet guide vanes. Electrical power is provided to the solenoid only when it is positioned to OFF. In case of electrical failure the system is fail-safe on.

Particle Separator - The particle separator kit (206-706-212-119) is equipped with a PART SEP purge switch located on the pilot's overhead console. With this switch in the ON position, engine bleed air is used to purge debris from the separator. With this switch in the OFF position, engine bleed air is not used to purge debris from the particle separator. However, there is still some performance loss due to restricted inlet airflow. Electrical power is provided to the solenoid only when it is positioned to OFF. In case of electrical failure the system is fail-safe on. Performance charts contained in the appropriate supplement provide data for use with this kit.

Hydraulic System Switch - The HYDRAULIC SYSTEM switch is located on the overhead console and controls the operation of the hydraulic bypass solenoid. The solenoid is electrically de-energized open when the hydraulic switch is in the ON position, allowing pressurized hydraulic fluid to flow into the hydraulic manifold. Electrical power is provided to the solenoid only when it is positioned to OFF. In case of electrical failure the system is fail-safe on.

Lighting Systems

The lighting system includes both interior and exterior lighting. The interior lighting system includes the cockpit map reading light, instrument and control panel lighting, and cabin lighting. The exterior lighting system includes position, anti-collision, and landing lights.

The cockpit map reading light features a narrow spotlight or wide floodlight beam. It is a multipurpose utility light designed to provide red or white illumination and is protected by its own circuit breaker. Controls for ON/OFF, BRIGHT/DIM, BLUE/WHITE, and SPOT/FLOOD are incorporated into the cockpit light body. 28 Vdc power is supplied through the CKPT LIGHTS circuit breaker located on the overhead console.

Instrumentation lighting 28 Vdc power is supplied through the INSTR LIGHTS circuit breaker located on the overhead console. The INSTR LT rheostat knob, also located on the overhead console, adjusts light intensity. Rotation of this knob operates the power OFF/BRT switch that provides power to both the 28 and 5 Vdc lighting systems. The INSTR LT rheostat also enables the CWAP BRT/DIM switch.

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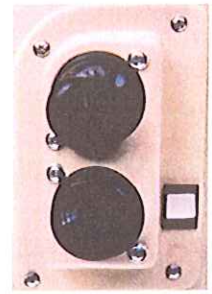
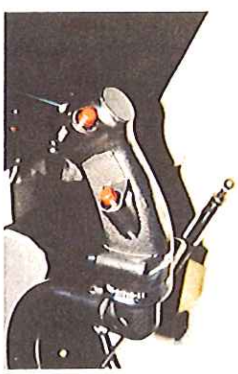
Bright/Dim Switch - With the INSTR LT rheostat adjusted out of the OFF position, the CWAP light intensity can be adjusted to either bright or dim intensity by positioning the BRT/DIM switch to the desired position. The CWAP lights will remain at the selected intensity until the INSTR LT rheostat is turned off, another switch position is selected, or power is removed from the DC bus. The ENGINE OUT, RPM, BATTERY HOT, ENGINE OVSPD, and FLOAT TEST warning and advisory lights will not dim.



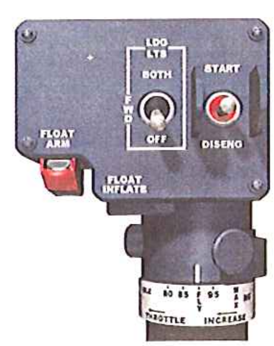
Interior Lighting System - Two individual white cabin lights provide passenger area illumination at the pilot's discretion. These two lights are protected by their own circuit breaker. They are regulated by the CABIN LT switch and by individual switches located near each light. When the CABIN LT switch is placed in the CABIN LT position, all four cabin lights illuminate. When the switch is placed in PASS position, the passenger can individually control each light.

Exterior Lighting System And Position Lights - The exterior lighting system includes position lights, tail light, anti-collision light, landing lights, and related wiring and components. Position lights are located on the outboard side of the auxiliary fins on the horizontal stabilizer and the lower cabin left and right side. The (white) tail light is located on the tail rotor gearbox fairing. The anti-collision light is mounted on the top portion of the vertical fin. Dual landing lights are located in the lower portion of the helicopter nose section.



Flight Control Switches

Cyclic Switches - A two position transmit switch is mounted on the cyclic grip. Depressing the switch to the first detent position will activate internal communication system (ICS) between the pilot and passengers. Depressing the switch to the second detent allows for the pilot to transmit on the radio selected on the audio panel.



Collective Switches - If the aircraft is equipped with floats, the collective control head will have a guarded FLOAT ARM switch on the lower left area, used to arm the floats. A FLOAT INFLATE push button is located to the right of the arming switch. On the upper surface of the collective head, there is a three position LDG LTS switch. Also, a START/DISENGage switch replaces the start push button. An indexed bezel has been added to facilitate the matching of the PLA throttle position to the engine Ng during the transition from FADEC automatic mode to manual mode.

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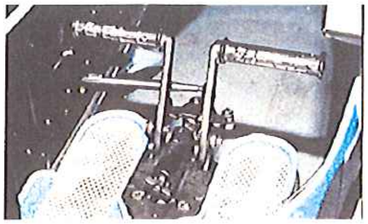
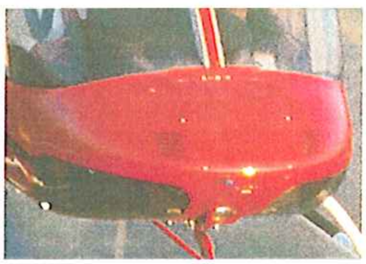


Tail Rotor Pedals And Adjuster - Tail rotor pedals can be positioned by means of the pedal adjuster to allow for individual comfort.

Ventilation System

Cockpit vent knobs - Air for cabin ventilation is obtained by opening the sliding window in each of the entrance doors. Additional ventilation to the cockpit may be obtained by pulling out the VENT control knobs and positioning the DEFOG blower switch on the overhead console to the DEFOG (ON) position.



Ram Air - When the VENT control knobs below the instrument panel are pulled out; ram air is forced into the air intake grills and directed through the plenum and flapper valve. The flapper valve assembly is opened or closed with the VENT control knob.

Defog - Two electrically driven axial flow blowers are installed in the inlet end of the defroster nozzles. The blowers are controlled by a DEFOG blower circuit breaker switch on the forward edge of the overhead console. The defog system is primarily used for ventilation and defogging during ground operation of the helicopter. When the system is used for ventilation or defogging, it is recommended that both VENT control knobs be extended to the full open position.

Seating

Crew And Passenger Seats - All seat back cushions are constructed of soft sponge with styrene backing. Seat cushions are constructed of soft urethane foam. Back and seat cushions are bonded to tubular frames and upholstered with fabric resistant to flame, fluids, and climatic conditions. A small storage area is located behind each crew seat. Aft facing seats are equipped with an adjustable headrest. (Refer to the placard in the side of the litter door).

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CREW COMPARTMENT

Restraint Assemblies

Each crew and passenger seat is equipped with a restraint assembly that consists of an inertial reel, shoulder harness, and an adjustable seat belt. The inertia reel is provided with an anti-rebound lock feature and is capable of retracting 22 inches of web belt.

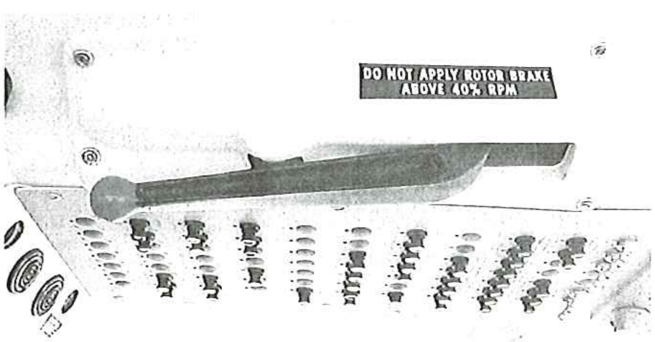
Flight Manual, Airworthiness Certificate, And Aircraft Registration Certificate Case

The Rotorcraft Flight Manual, used for operation of the helicopter, is located in the crew compartment. The aircraft document case (i.e. airworthiness and registration certificates) is riveted to the right forward side of the instrument panel console. These certificates are required to be carried in the aircraft at all times.

Hand Fire Extinguisher

A manually operated fire extinguisher is furnished with each helicopter. The extinguisher is located in the crew compartment between the pilot and co-pilot seats. A mounting bracket is the quick opening type for rapid removal of the extinguisher.

Rotor Brake



If a rotor brake is installed, application is limited to ground operation after the engine has been shut down and Nr has decreased to 40% or lower. Engine starts with the rotor brake engaged are prohibited. Do not stop the blades at the 12 o'clock and 6 o'clock position, as hot exhaust may cause damage to the blade above.

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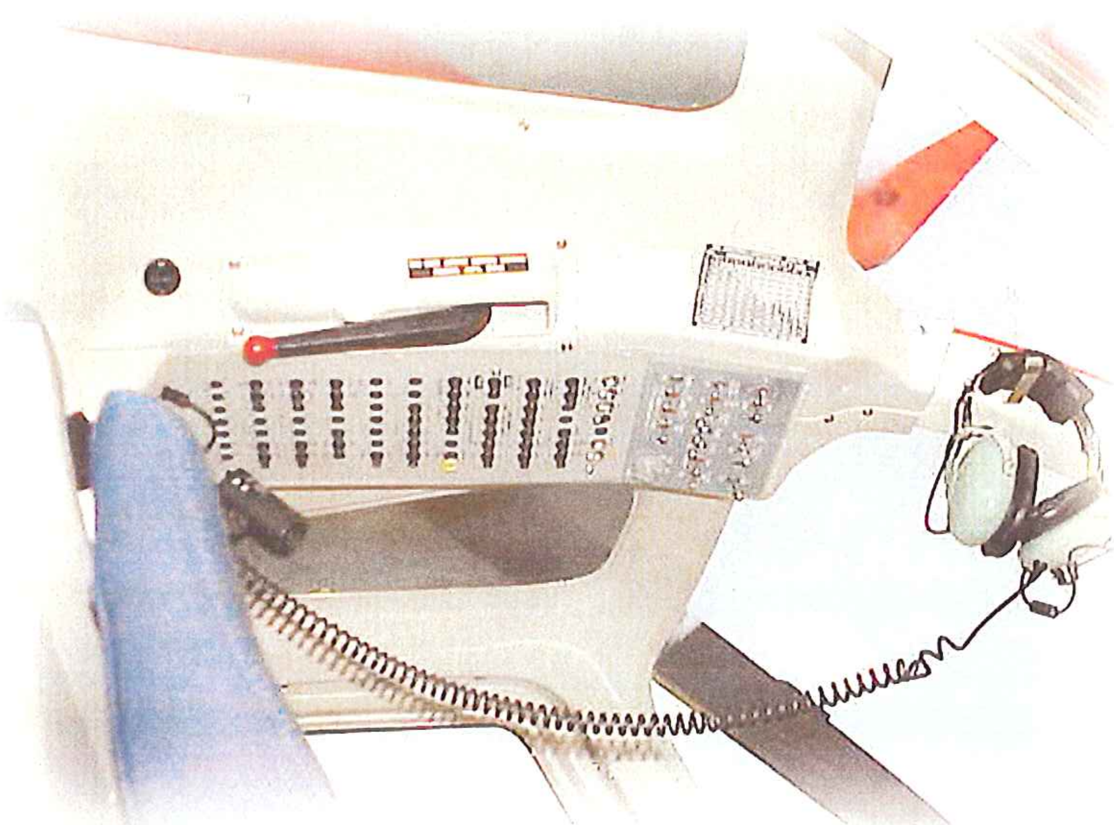
Electrical System



Training Academy

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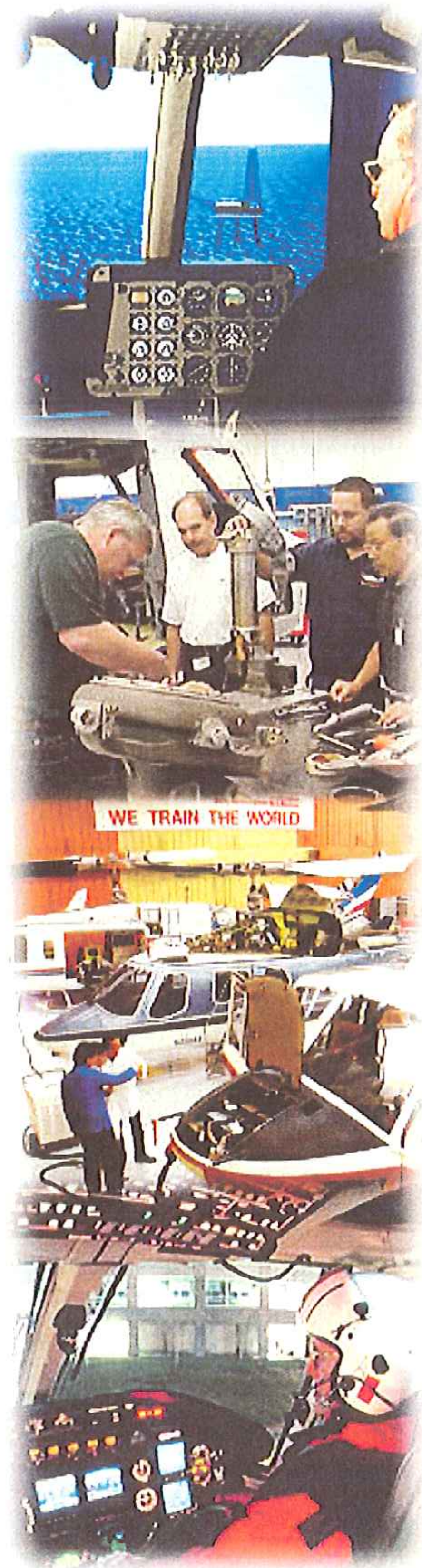

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ELECTRICAL SYSTEM

General

The helicopter is equipped with a 28 VDC electrical system. Aircraft electrical power is provided by a nickel-cadmium battery, a starter/generator and/or an external power receptacle.

Controls for the electrical system are located on the overhead console and instrument panel. Control relays, power relays, voltage regulator, and circuit breakers control and regulate the voltage transfer. Malfunction monitoring circuits are located in individual compartments.

The major components of the DC power system include the battery, starter/generator, DC control unit/voltage regulator, relays, 28VDC bus, and circuit breakers. All circuits in the electrical system are single wire with fuselage common ground return.

Battery System

The battery system includes the battery, a battery relay, and a battery switch with related wiring. The battery may be a nickel-cadmium 24 volt 17 ampere-hour battery or an optional 24 volt 28 ampere-hour battery. The battery is located in the nose of the aircraft. The battery relay is located below the battery in the landing light compartment and controls battery current to the main bus bar. This relay is actuated by the battery switch located in the overhead console, which opens and closes the circuit to the relay energizing coil.

External Power System

The external power system includes the external power receptacle, external power relay, and related wiring. The external power receptacle, located in lower front center of the nose section, is a polarized receptacle used as a contact point for external power plug-in.

The external power relay is located in the nose below the battery in the landing light compartment. The relay is an electrically operated switch between the external power receptacle and the main bus bar. It is controlled through the small positive pin from the external power source that energizes the circuit to the activating coil of the relay.

The 28 VDC Ground Power Unit (GPU) shall be limited to 500 amperes or less to reduce the risk of starter damage due to excessive heat.

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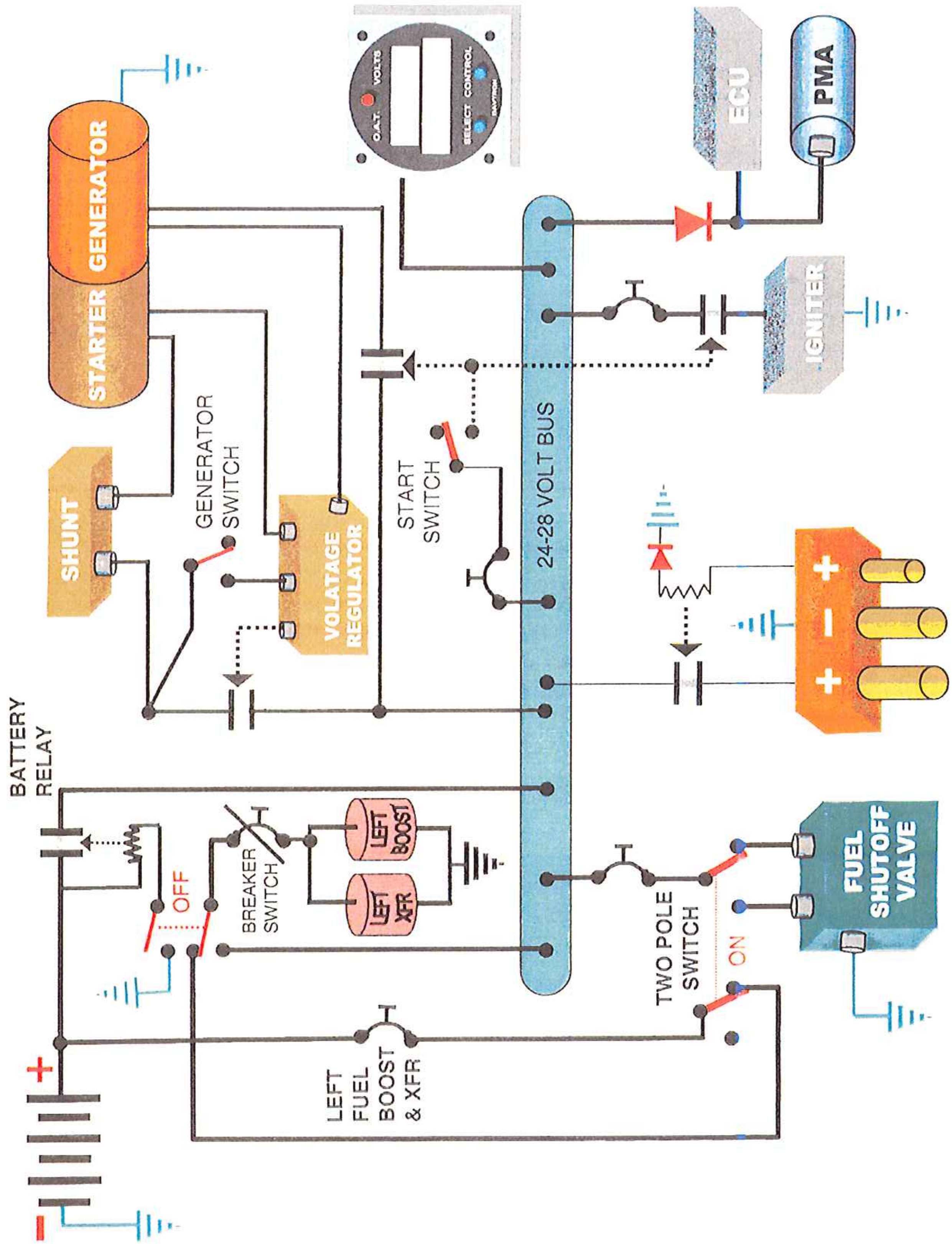
ELECTRICAL SYSTEM



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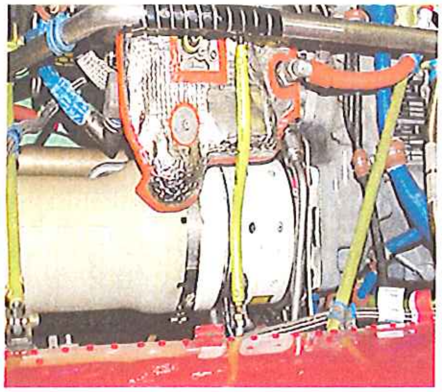
ELECTRICAL SYSTEM

Generator System

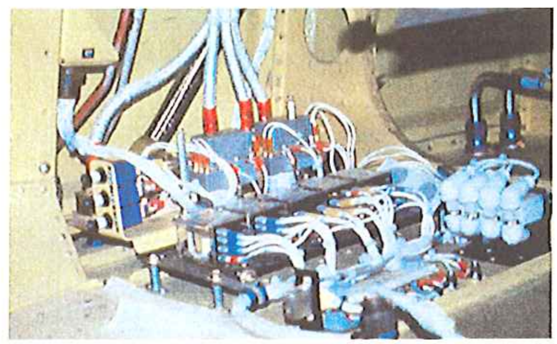


The generator system includes the generator portion of the starter/generator, generator control unit (GCU), line control relay, generator reset switch, and generator shunt.

The generator is located on the lower right aft side of the accessory gearbox and furnishes power at a regulated voltage for all dc electrical components on the helicopter. The generator switches onto the main bus when the generator switch is on and generated voltage exceeds the voltage on the bus by 0.30 to 0.42 volts.

The generator control unit controls the operation of the electrical system. It operates the line control relay, regulates generator voltage, and provides protection against overvoltage and reverse current. The GCU contains an electronic voltage regulator to control voltage output of the dc generator and a circuit to energize the line contractor when correct conditions exist.

The line control relay is installed on the electrical panel assembly located on the equipment shelf above the baggage compartment. It opens or closes the power circuit between the dc bus and the generator and is controlled by the presence or absence of a proper output voltage from the generator control unit.



The generator reset switch, located on the overhead console, is a double-pole, triple throw, spring-loaded switch with momentary contact in the RESET position. It completes the generator field circuit in the ON position, supplies voltage to reset the generator field circuit in the ON position, and disconnects generator field circuit in the center OFF position.

The generator shunt is installed on the electrical panel assembly on the equipment shelf above the baggage compartment and provides a voltage drop proportional to the current to operate the loadmeter.

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ELECTRICAL SYSTEM

Starter/Igniter System

The starter/igniter system includes the starter portion of the starter/generator, starter relay, field/igniter relay, igniter, and starter switch.

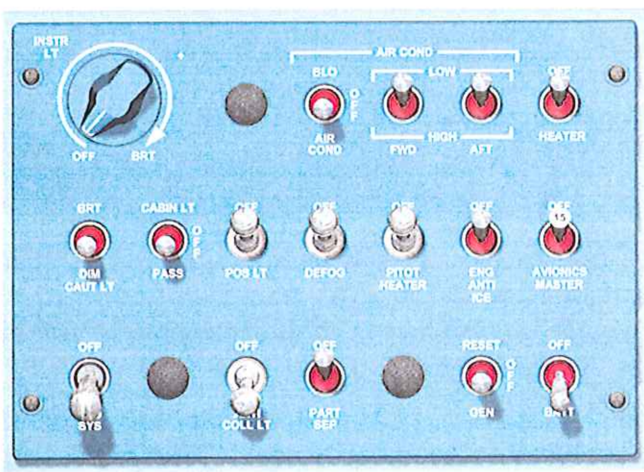
The starter/generator is energized by the starter relay to start the engine.

The starter relay is installed on the electrical panel assembly located on the electrical shelf above the baggage compartment. Direct current is supplied to the starter through the starter relay when the start switch is engaged.

The field/igniter relay is installed on the electrical panel assembly located on the electrical shelf above the baggage compartment. Direct current is supplied to the igniter and start field suppress section of the GCU through the field/igniter relay when the starter is engaged.

The tension capacitor discharge ignition exciter (igniter exciter) is located on the lower left section of the engine accessory case. This exciter provides increased voltage to the igniter plug to insure fuel ignition during the engine start cycle.

Circuit Breakers and Switches



Circuit breakers and switches are mounted in the overhead console and instrument panel. The circuit breakers are mounted on the overhead console. Circuits can be separated and closed by operating these push/pull circuit breakers.

Battery Switch – The battery switch is installed on the overhead console and controls the battery relay that connects the battery to the D.C. bus. The switch positions are OFF and BATT.

Start Switch – The start switch is located on the collective switch box. It contains two sets of spring loaded contacts that provide momentary contact in either the START or DISENG positions. When the switch is positioned to start or disengage only one set of contacts move.

Generator Switch – The generator switch is installed on the overhead console and controls generator output by opening and closing the generator field circuit. The switch is a double pole, double throw, spring loaded design with only momentary contact in the RESET position. The switch positions are GEN, OFF, and RESET.

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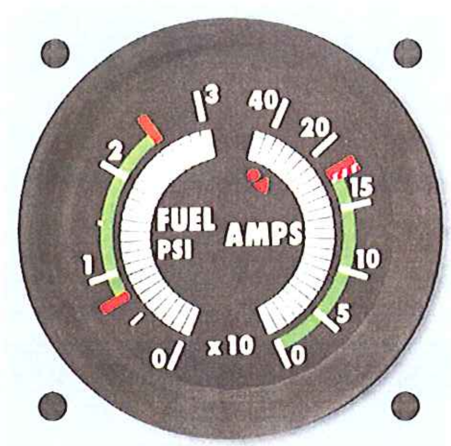
ELECTRICAL SYSTEM

Electrical System Indicators

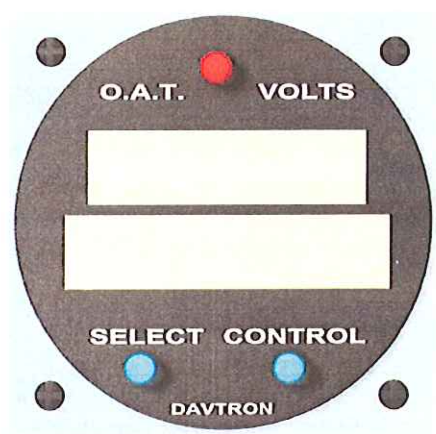
Electrical system indicators include a DC ammeter, voltmeter, BATTERY HOT light, BATTERY RELAY light, START light and a GEN FAIL light.



Fuel Pressure / DC Ammeter – The fuel pressure / ammeter gage is a dual display instrument. The ammeter indicates the load in amperes that is being supplied to the 28 VDC bus by the generator. The ammeter side of the indicator is powered by its own circuit breaker.

Ammeter 400 Amp Max Scale – The ammeter indications will be continuous regardless of load. The indicator will not drop to zero if limits are exceeded. To insure limits are not exceeded, the pilot may switch the generator OFF prior to a generator exceedance. Following a brief time, generator switch can be positioned to GEN (on).

Voltmeter – The voltmeter is included in a multifunction indicator mounted in the upper left area of the instrument panel. The indicator also displays Outside Air Temperature and Clock functions. A button located on the center top of the instrument changes the top display between Volts (e.g. 28E) and OAT (Celsius and Fahrenheit). When power is applied to the instrument the display defaults to the voltmeter reading. The Voltmeter display receives its power from the 28 VDC bus through the OAT / V INSTR circuit breaker. The 28 VDC bus through this circuit breaker is also the source of the voltage displayed by the Voltmeter. When all electrical power is turned off, the Voltmeter display disappears.



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ELECTRICAL SYSTEM

Battery Hot Annunciator – The BATTERY HOT light is utilized with either the basic ship 17 amp/hour or the optional 28 amp/hour battery. The 17 amp/hour battery incorporates two thermal switches while the 28 amp/hour battery incorporates three thermal switches. While both of the 17 amp/hour battery thermal switches are used in the BATTERY HOT light circuit, only two of the three thermal switches available on the 28 amp/hour battery are used.

The 17 amp/hour battery thermal switches will close at a temperature of 145 ± 5°F (62.7 ± 2.8°C). The 28 amp/hour battery switches will close at a temperature of 160 ± 5°F (71.1± 2.8°C). If any of the thermal switches close, the BATTERY HOT light will illuminate.

Battery Relay Annunciator – The BATTERY RELAY light will illuminate if the battery relay has remained in the closed (energized) position after the battery switch has been set to OFF. If the battery relay remains energized after the battery switch has been set to OFF, battery power will remain on the 28 VDC bus. This will power the Caution, Warning, and Advisory Panel (CWAP) to allow illumination of the BATTERY RELAY light even if the generator is OFF.

Start Annunciator – The START light will be illuminated when the starter relay is energized. The starter relay will be energized when the start switch is positioned as follows:

> With the FADEC MODE switch positioned to AUTO, the starter relay will stay engaged until the gas producer (Ng) speed reaches 50 ±1%.
>
> With the FADEC MODE switch positioned to MAN, the starter relay will stay engaged until the start switch is released from the START position.

Generator Fail Annunciator – The GEN FAIL light is controlled by generator relay. The light will illuminate when generator relay is de-energized and not connecting the generator output to the D.C. bus. The generator relay is energized by the generator control unit / voltage regulator when generator output climbs through a threshold of 24 ± 2.4 VDC. Prior to the generator relay being energized, the GEN FAIL light is on. Once the generator relay is energized the GEN FAIL light is extinguished.

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Fuel System

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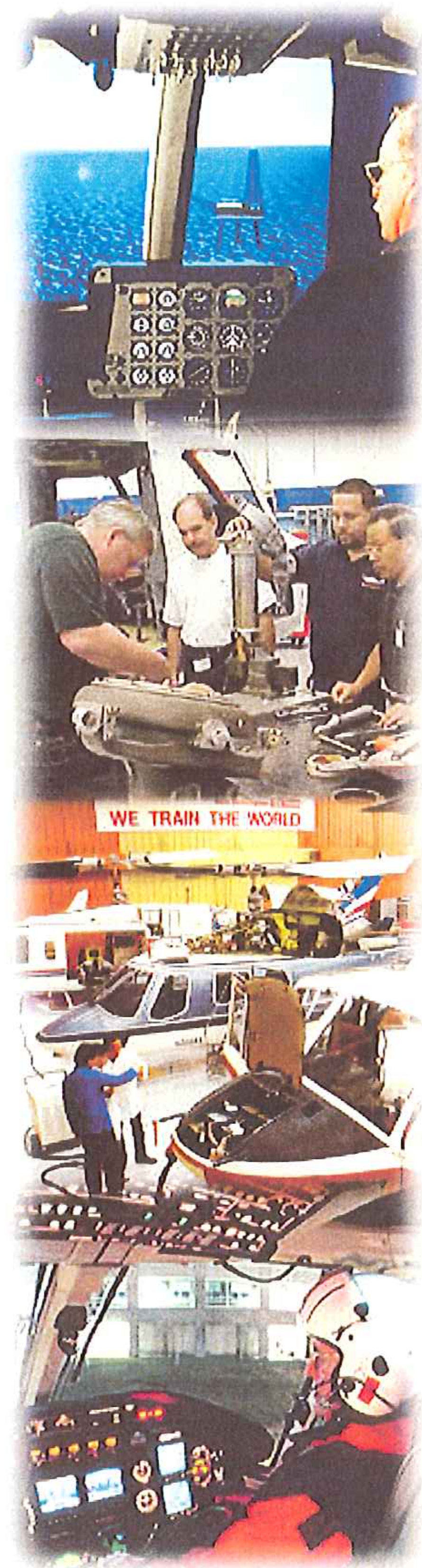

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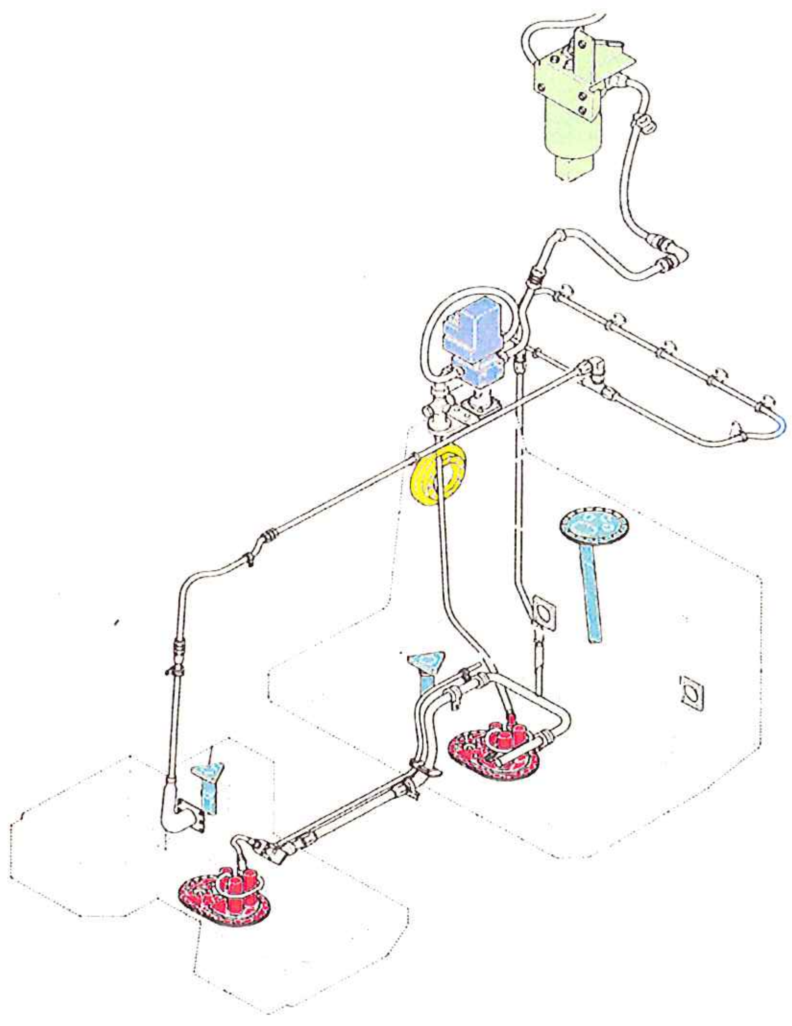
FUEL SYSTEM

System Description

The basic Model 407 fuel system consists of two fuel cells, pumps for transferring fuel between the two cells, pumps for supplying fuel to the engine, a fuel cell venting system, a fuel cell quantity indicating and low level detection system. Each fuel cell has an electrically operated fuel drain solenoid valve. An electrically operated fuel shut off valve is located at the top of the aft fuel cell. Several caution lights indicate abnormal operation of their related component. Fuel is transferred between the aft and forward cells by two means: a gravity transfer line permits fuel to transfer from the aft to forward cell whenever the level of fuel in the aft cell is above the height of the standpipe, and a pressure transfer line from the forward transfer pumps to the aft (Main) tank. The pressure and transfer systems each have two electrical pumps mounted on a common sump plate in the bottom of the forward cell. Each pair of pumps is equipped with check valves and pressure switches.



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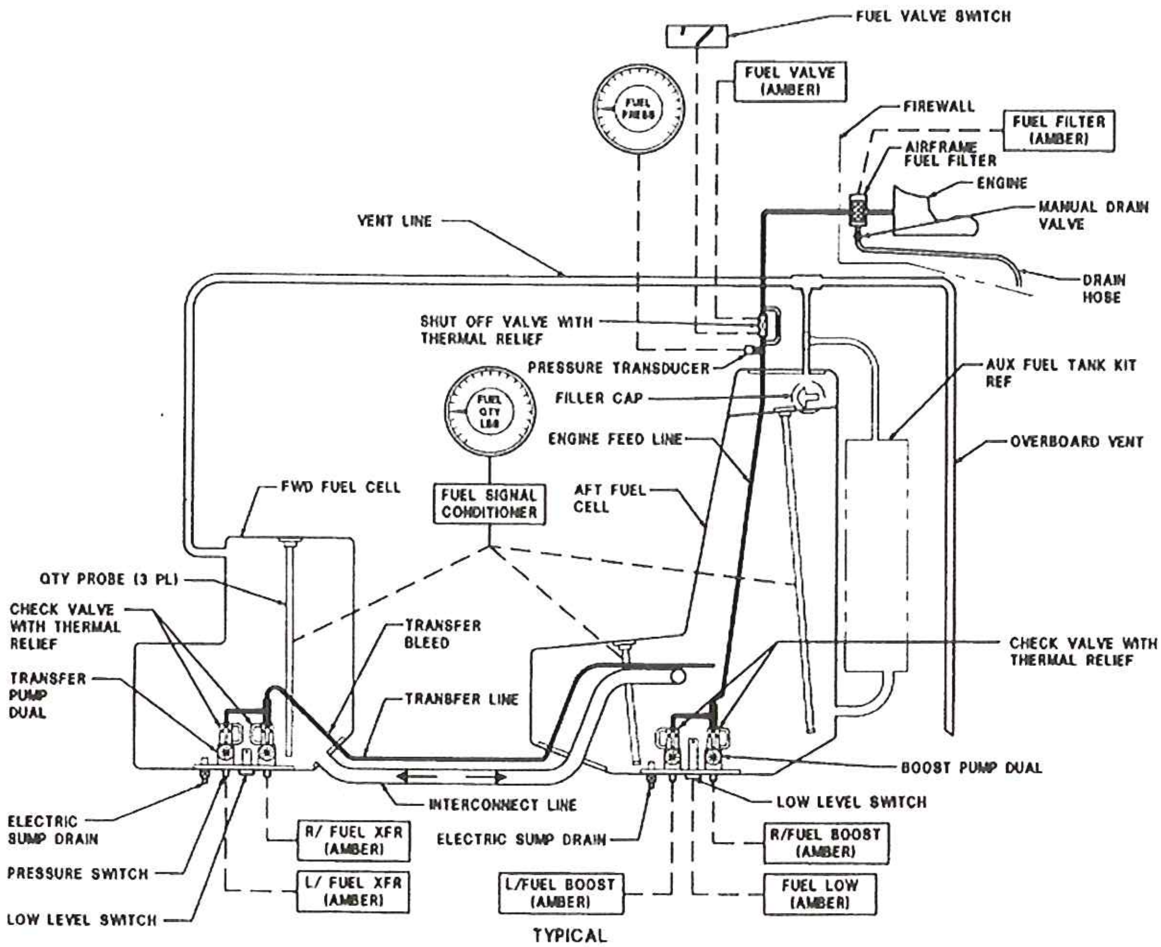
FUEL SYSTEM



System Components

Fuel Cells – The basic aircraft has two crash resistant bladder type fuel cells, which are serviced through a filler port located on the right side of the aircraft. These tanks have a combined capacity of 130 US gallons (127.8 US gallons usable). The aft cell is located underneath and behind the aft passenger seats and has a total capacity of 92 gallons (348 Liters). The forward cell is located underneath and between the middle passenger seats and has a capacity of 38 gallons (144 Liters). Both tanks are connected by gravity and return lines on the floor of the main cabin that are covered by a protective fiberglass cover.

Auxiliary Fuel Cell – Every aircraft has provisions for an optional 20 gallon U.S. (76 Liter) auxiliary fuel tank that can be installed in the baggage compartment. The auxiliary tank has no fuel quantity probe or fuel pump. The auxiliary tank is mounted in such a way that the tank itself and the fuel do not add any additional weight to the baggage floor. Installation of the AUX TANK does not reduce the baggage compartment maximum load of 250 pounds.

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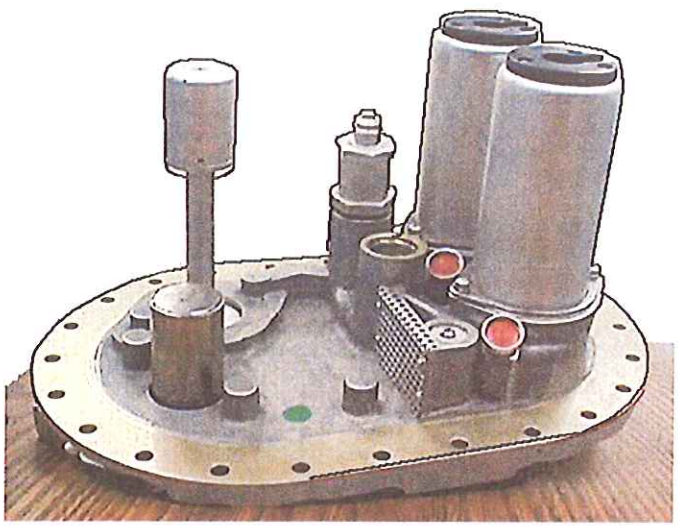
FUEL SYSTEM

Forward Fuel Transfer Pumps – Two transfer pumps mounted on a common sump plate are located on the floor of the tank which will transfer all usable fuel to the aft cell. Each pump has a pressure switch that controls a transfer pump caution light on the CWAP. With the pump switch on and the transfer pump producing normal pressure, the related caution light will be extinguished indicating proper operation. If a transfer pump fails, the pressure switch will sense low pressure and cause the respective FUEL XFR caution light to illuminate on the CWAP. Each transfer pump sends fuel through a one way check valve located at the outlet of each pump. This check valve will ensure that if either transfer pump becomes inoperative, fuel will be pumped to the aft tank and not through the inoperative pump back into the forward fuel tank. Individual fuel lines are then connected to a 'T' fitting that connects the supply from each pump to a single line.



Fuel Boost Pumps – Two boost pumps mounted on a common sump plate are located on the floor of the aft cell. Each individual pump is capable of lifting all usable fuel past the fuel pressure transducer, through the fuel valve, then through two filters and on to the HMU high pressure pump. Each boost pump has a pressure switch that controls a boost pump caution light on the CWAP. With the pump switch on and boost pump operating normally, the related caution light should be extinguished. Each boost pump sends fuel through a one-way check valve located at the outlet of each pump. This check valve will ensure that if either boost pump becomes inoperative, fuel will be pumped to the engine and not through the inoperative pump back into the aft fuel tank. Individual fuel lines are then connected to a 'T' fitting that connects the supply from each pump to a single line.

Fuel Sump Drain Valves – Fuel drain switches provide a means of draining fuel from both the forward and aft cells. Sump drain valves are located on the fuel boost and transfer pump sump plates and are electrically operated by means of the drain switches located aft of the right-side passenger door. Drain switches are wired through the fuel valve switch on the instrument panel. The fuel valve switch must be in the OFF position to operate either fuel cell drain valve.

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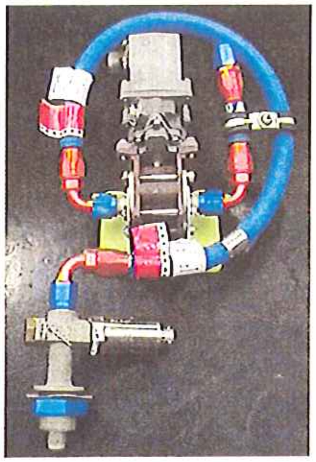
FUEL SYSTEM

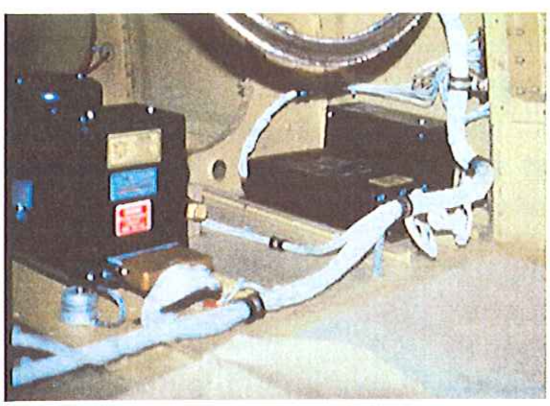
Fuel Pressure Transducer – Transducers convert a wet line (fluid passing thru tubing) signal into an electrical signal. Aft cell boost pump fuel pressure is displayed in PSI (pounds per square inch) on the Fuel and Amp Gage by means of a fuel pressure transducer.



Fuel Valve – The fuel valve is located behind the aircraft right side body fairing above the refueling port and is electrically controlled by a switch located on the lower right corner of the instrument panel. A fuel valve caution light will illuminate when the fuel valve is in transit or has stopped somewhere between the full OPEN or full CLOSED position.

Fuel Filter – The airframe fuel filter is located on the forward firewall on the right side of the power plant. It is equipped with a built in filter bypass valve and impending bypass switch that is electrically connected to an amber caution light on the CWAP. At .875 ± .125 psid (differential fuel pressure) the airframe filter caution light will illuminate. The filter will bypass at 3.75 ±25 psid, however there are no further cockpit indications. A red press-to-test button is located on the top of the filter to check the electrical connection to the fuel filter caution light.

Fuel Quantity System



A signal conditioner is located on the electrical shelf behind the rear forward facing seat. Three separate fuel quantity probes (one located in the forward tank and two located in the aft tank) send signals indicating fuel quantity (based on fuel levels and density) to the signal conditioner. A microprocessor in the signal conditioner uses the information provided by the three fuel probes to compute the weight of the fuel, and displays the fuel quantity in pounds on the fuel quantity gage. The fuel signal conditioner senses the auxiliary tank's presence by a micro switch in the baggage compartment. When the AUX TANK is installed the signal conditioner recalculates fuel quantity in the main (aft) tank, including the additional 20 gallons in the auxiliary tank using the aft fuel capacity probe. The combined total quantity of fuel is then displayed on the fuel gage. With the AUX TANK installed fuel

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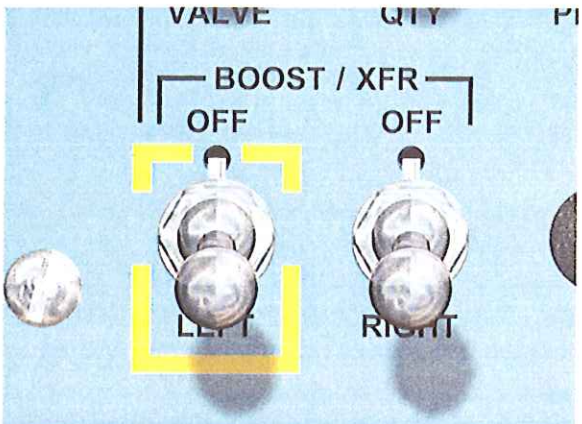
capacity is increased to 150 Gallons U.S. (567 Liters). The signal conditioner also activates the fuel low caution light should the fuel quantity in the aft tank be at or less than 110 ± 15 pounds.

The signal conditioner carries out a four second power Built-In-Test (BIT) (there is no connection between the signal conditioner and the FADEC BIT) when the unit is first provided power. If a failure is detected during the BIT the signal conditioner will blank the fuel quantity gage. The signal conditioner also performs a continuous BIT whenever the unit is powered. If an error is found during the continuous BIT, or a fuel probe is malfunctioning, the signal conditioner will blank the fuel quantity gage display.

Fuel Low Light – A float type low-level detector is mounted on the aft fuel cell sump plate and is calibrated to send a signal to the Signal Conditioner that will activate a Low Fuel caution light on the CWAP when approximately 110 ±15 pounds of fuel remain in the AFT tank.



LEFT and RIGHT BOOST / XFR switch – The left and right fuel boost / XFR (transfer) switches are a circuit breaker toggle design and are located on the overhead console. Each switch powers a set of pumps. The left boost / XFR pump switch controls the left boost pump in the aft fuel cell and the left transfer pump in the forward cell. The right boost / XFR pump switch controls the right boost pump in the aft cell and the right transfer pump in the forward cell. In normal ground run and in-flight conditions the aircraft battery switch is in the ON position. DC power (battery and generator) is connected to the single aircraft bus powering all items wired to the bus, including the left boost and transfer pumps. In the event the battery switch is positioned to OFF during aircraft operations (fuel valve on), an alternate circuit provides electrical power directly from the battery to the left boost and left transfer pumps.

The LEFT FUEL / XFR switch is outlined by a yellow border to identify that it has an alternate circuit. In the event that a short circuit or battery hot condition occurs in the helicopter and all DC bus power (battery and generator) are shut off, it is desirable to maintain the operation of one transfer and one boost pump, allowing access to all fuel in the fuel cells. To enable this feature, the fuel valve switch must be positioned to ON, the left fuel boost / XFR circuit breaker switch must be ON, and the battery switch must be turned OFF.

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Fuel Transfer Lights

When the forward fuel cell is nearing depletion the signal conditioner uses programmed time delays to control the transfer pumps and lights. The time delay allows continued operation of the transfer pumps to ensure all of the fuel in the forward tank is transferred to the aft main tank. When the forward fuel cell is empty there is approximately 193 ±15 pounds of fuel remaining in the rear fuel cell.

S/N 53000 THRU 53174

When the forward fuel cell is nearing depletion the signal conditioner uses time delays of 2½ minutes to control the transfer pumps and caution lights. The time delay allows continued operation of the transfer pumps to ensure all of the fuel in the forward tank is transferred to the main tank. The caution lights remain operational during the time delay and will illuminate if pump pressure drops. Once the time delay periods are ended, the transfer pumps and the L/FUEL XFR and R/Fuel XFR light circuits are deactivated. The transfer pump and light circuits will stay inoperative as long as the forward fuel cell is empty.

S/N 53175 AND SUBSEQUENT

Activation of either annunciator circuit is dependent on the relationship between the fuel quantity in the forward fuel tank and the total quantity of the fuel system.

(Refer to the table on the following page). When the fuel signal conditioner detects a forward fuel quantity of less that 25 pounds and a total quantity of less than 250 pounds for 10 consecutive seconds, L/FUEL XFR or R/FUEL XFR transfer lights are deactivated to prevent intermittent light flickering while remaining fuel is transferred from the forward tank to the aft tank. When the forward tank probe and the low level detector sense a low fuel condition, input signals to the fuel conditioner are removed. With input signals removed, the fuel signal conditioner utilizes a 360 second time delay prior to disabling the caution light. The pumps continue running for 360 seconds to ensure all forward tank fuel is transferred to the main fuel tank. The annunciator and transfer pump circuits will stay inoperative until the fuel system is refueled.

The L/FUEL or R/FUEL XFR transfer light will illuminate when their respective fuel pressure switch senses a decreasing boost pump pressure of 1.5 psi ± 0.5 psi, provided the conditions shown in table below are met.

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TRANSFER LIGHT ACTIVATION TABLE

FWD TANK <25 POUNDS FOR 10 CONSECUTIVE SECONDS	TOTAL FUEL<250 POUNDS FOR 10 CONSECUTIVE SECONDS	L/FUEL XFR OR R/FUEL XFR LIGHT ILLUMINATED
FALSE	FALSE	YES
FALSE	TRUE	YES
TRUE	FALSE	YES
TRUE	TRUE	NO

Fuel Drain Procedures

(Excerpt from the Model 407 Flight Manual)

FUEL SUMP

Drain fuel sample as follows:

a. Left and Right FUEL BOOST/XFR circuit breaker switches - OFF

b. BATT switch — BATT (ON).

c. FUEL VALVE switch — OFF.

d. PUSH the FUEL CELL DRAIN button(s), (on the right aft side of the aircraft fuselage) — press, drain sample, then release.

A/F Fuel filter

Drain and check before first flight of the day as follows:

a. FUEL VALVE switch — ON.

b. Left and Right FUEL BOOST/XFR circuit breaker switches — ON.

c. Fuel Filter Drain Valve — Open, drain sample, then close.

NOTE

Filter test button is located on top of fuel filter.

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Fuel filter test button — Press and check A/F FUEL FILTER caution light illuminated. Release switch and check light extinguished.

FUEL VALVE switch — OFF.

BAT switch — OFF

Fuel Flow and Charts

Fuel flow charts present aircraft fuel consumption during level flight as a function of altitude, OAT, airspeed, and gross weight. These charts are based on estimates and limited flight test data. They are applicable to the basic helicopter with all doors installed and without any optional equipment which would appreciably affect lift, drag, or power available. This data does not include effects of bleed air heater, ECS (air-conditioning), particle separator purge, or anti-ice operation on fuel consumption. Also, fuel consumption may vary between engines under the same operating conditions. It is recommended that the operator conduct fuel consumption checks to adjust the presented data as necessary.

The expanded performance section of the Manufacturer's Data, provides Fuel Flow vs. Airspeed charts ranging from sea level to pressure altitude 17,000 feet

EXAMPLE:

Problem: What is the fuel burn for 4600 pounds, at 4000 feet pressure altitude, and 7° C, at 130 knots indicated airspeed.

Solution: Find the appropriate chart, and begin by locating the indicated airspeed of 130 knots at the bottom of the chart. Locate the aircraft weight of 4600 pounds (lower left of chart center), then follow the curved line to the right and upwards till you intersect the appropriate indicated airspeed line of 130 knots. Proceed horizontally to the LEFT until you intersect the FUEL FLOW LB/HR (fuel flow pound per hour) of approximately 335 lbs. per hr. You may continue further LEFT and read the approximate aircraft torque for these conditions.

Long Range Cruise (LRC) – The solid line labeled LRC indicates the optimum long-range cruise airspeed for best fuel economy. LRC is not depicted where it would occur above the continuous transmission limit line.

The long range cruise airspeed for 4600 pounds is found the same way as above, however enter the appropriate chart at the aircraft gross weight of 4600 pounds and then proceed along the curved line to the right and upwards, till you intersect the LRC line. Proceed DOWN vertically to the bottom of the chart and read the aircraft indicated (118 knots) or true airspeed (120 knots) for these conditions.

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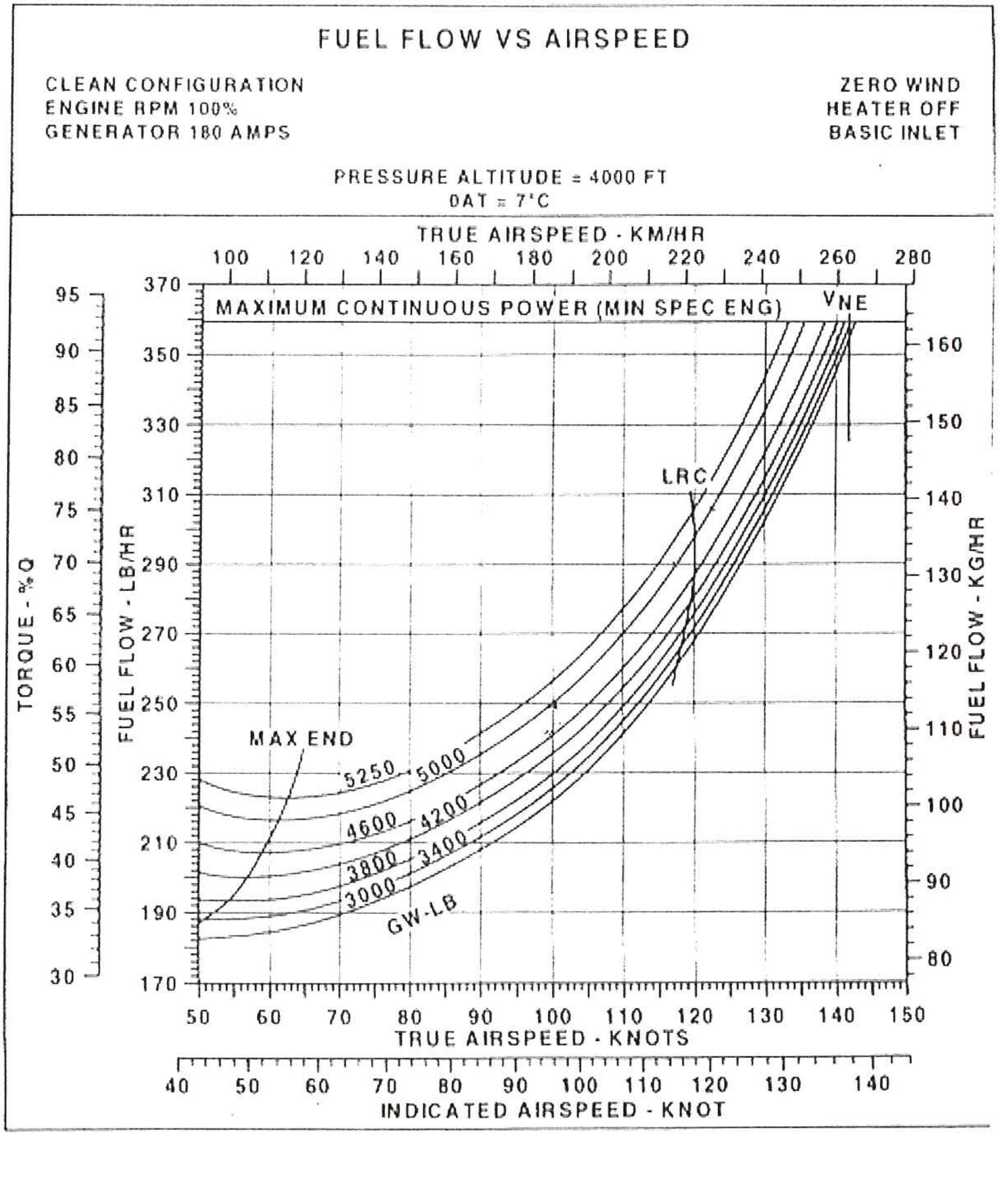
Maximum Endurance (Max End) – The solid line labeled Max End indicates the aircraft gross weight and airspeed relationship that would yield the minimum fuel burn and give the aircraft the most time aloft.



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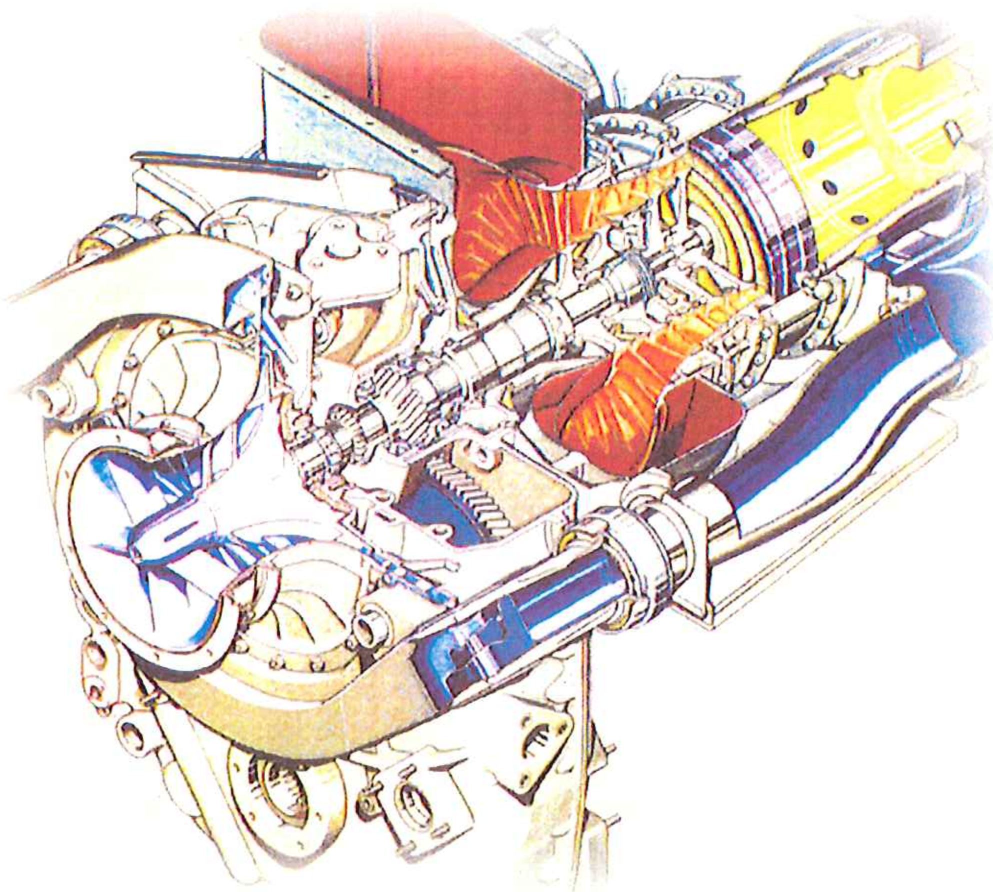
Powerplant



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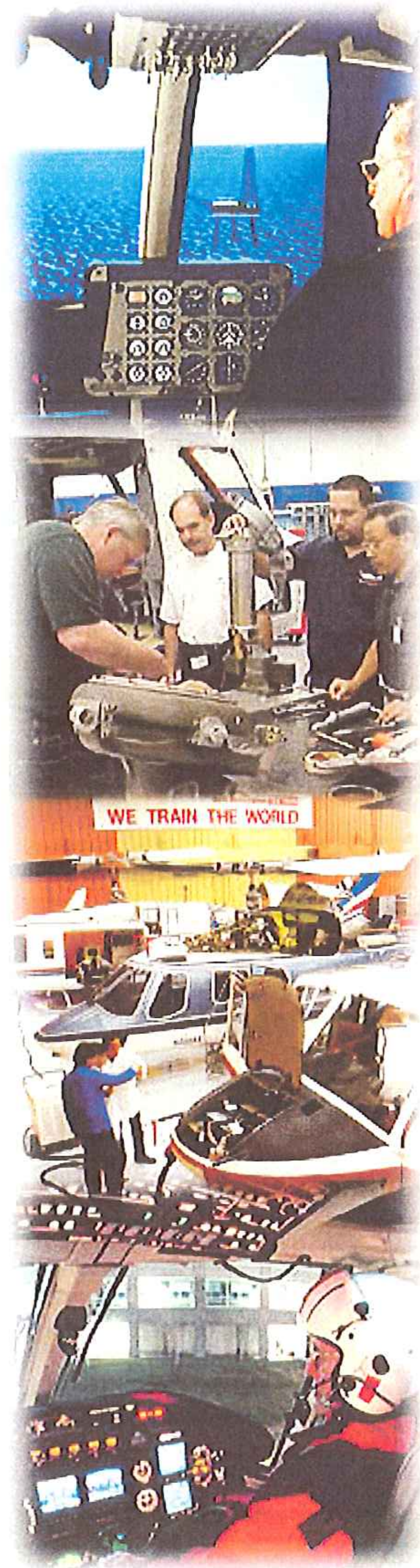

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POWERPLANT

General

The Rolls Royce 250/C47B engine is an internal combustion turboshaft engine featuring a free turbine. The gas generator is composed of a single stage, single entry centrifugal flow compressor directly coupled to a two-stage gas generator turbine. The power turbine is a two- stage free turbine which is gas coupled to the gas generator turbine. The integral reduction gearbox has front and rear drive splines to mate with aircraft drives. The engine has a single combustion chamber with single ignition. The output shaft centerline is located below the centerline of the engine rotor and the single exhaust outlet is directed upward. The engine incorporates a Full Authority Digital Electronic Control (FADEC) system.

The Model 250/C47B is a thermodynamically rated 813 shaft horsepower (SHP) engine. To provide reserve power for high altitude and hot day performance, the engine torque has been derated through the Hydro-Mechanical Unit (HMU) or fuel control to 674 SHP for takeoff and 630 SHP for continuous operation.

To further clarify the 250/C47B's engine horsepower, the engine dataplate identifies the engine's rated horsepower as 650. This rated horsepower is what Allison guarantees the engine will provide at sea level at a specific fuel flow and MGT. This ensures all 250/C47B engines sold to Bell Helicopter meet or exceed the rated horsepower specification. Therefore, the rated horsepower shown on the engine dataplate is not the maximum horsepower that the engine will deliver when installed in the 407 helicopter.

The engine is horizontally mounted aft of the transmission and above the fuselage to simplify the drive system, improve the inlet/exhaust arrangement, and to reduce cabin noise. To provide better structural integrity, it is supported by three bipod and one horizontal mount on the service deck. The transmission is coupled to the engine by means of the main drive shaft and freewheeling unit.

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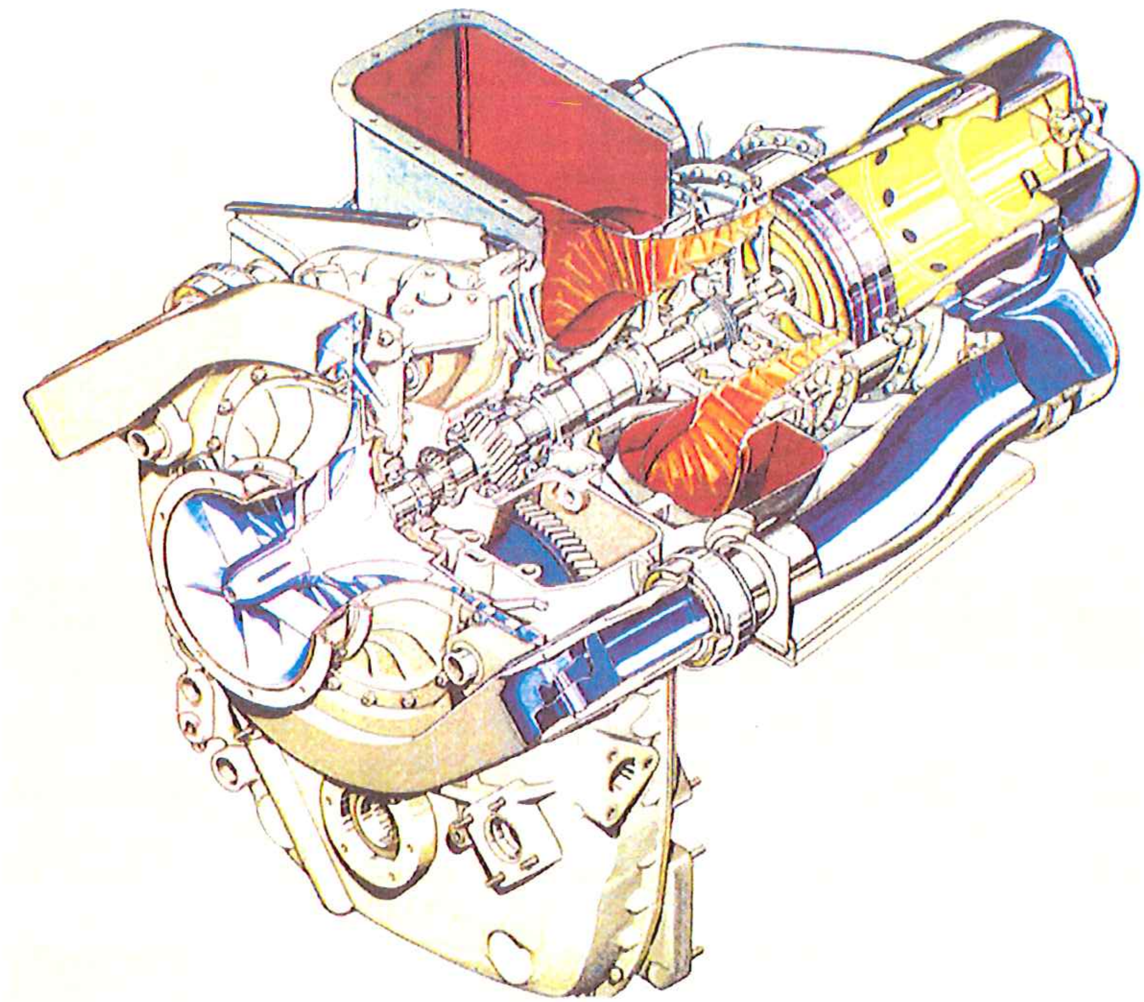
POWERPLANT

The major engine components are a compressor, combustion section, turbine section, and power and accessory gearbox.



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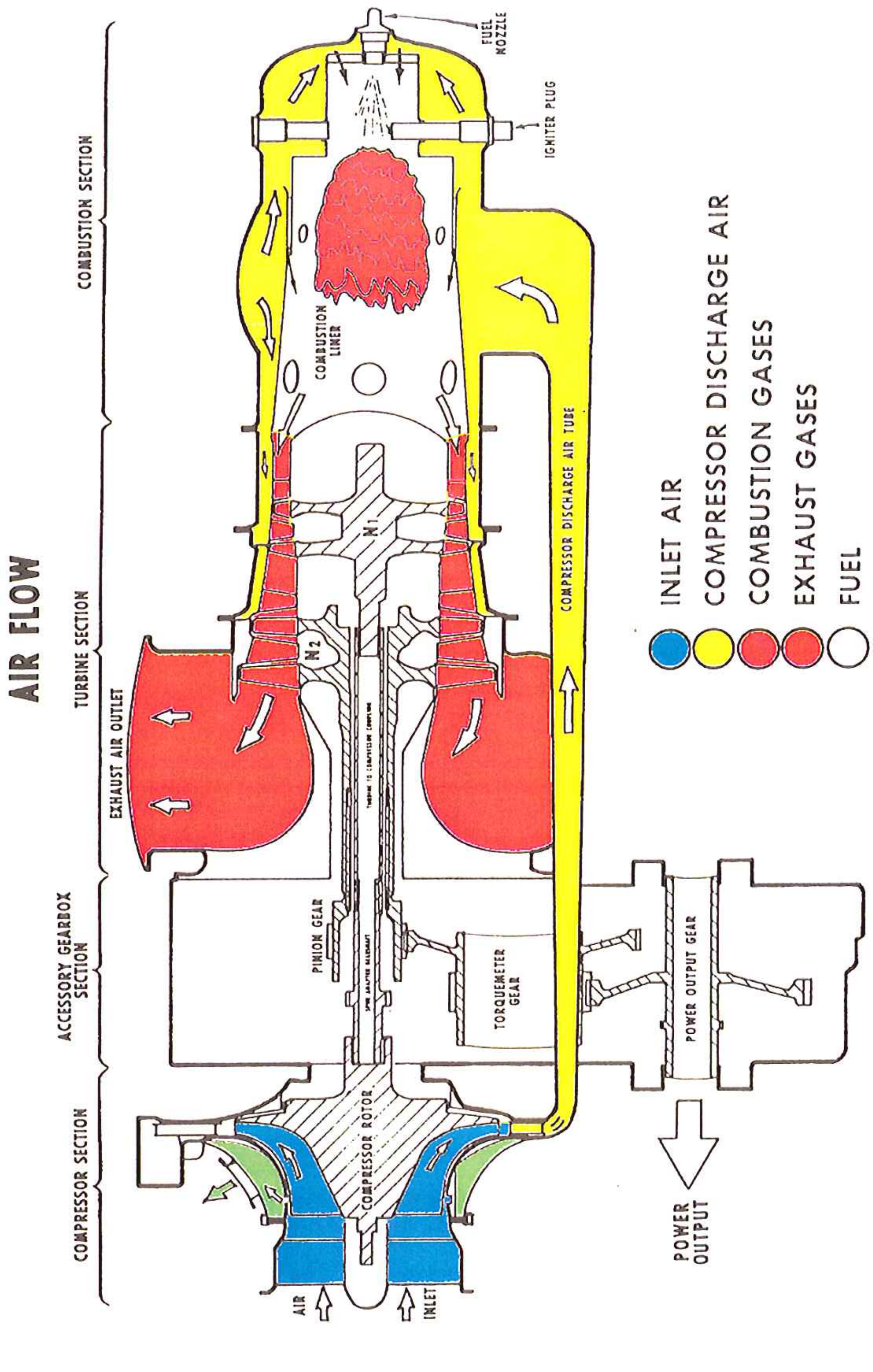

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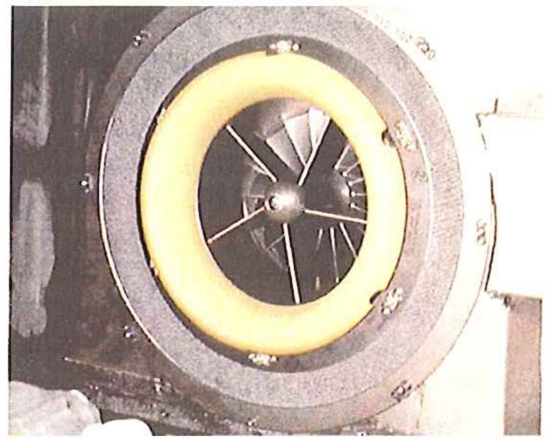
POWERPLANT

Compressor

The compressor assembly consists of a compressor front support, shroud housing, diffuser, rear support assembly, centrifugal impeller, scroll assembly, mount assembly and bearings. The front support encloses the front bearing and supports it through five hollow inlet struts. The struts are hollow to provide anti-icing using compressor discharge air and to permit introduction of oil to and from the compressor front bearing. The compressor rear bearing is mounted in the rear support assembly and is lubricated from the gearbox.



The compressor impeller is machined from a single piece of forged titanium. The vanes transition from axial to centrifugal, eliminating the need for stators. At 100% Ng the compressor is rotating at 51,000 RPM. The compression ratio is approximately 9.2:1 (9.2 Bars or 133 PSI). This rapid compression will increase the temperature of the compressed air approximately 291°C (555°F).

The compressor liner has a narrow slot machined into the perimeter near the front. This slot is known as the inducer bleed port. At idle RPM, a portion of the airflow will vent into the atmosphere, unloading the compressor, allowing for more rapid acceleration.

At higher RPM when maximum engine efficiency is required, atmospheric air will enter the compressor, increasing the volume of air entering the engine.

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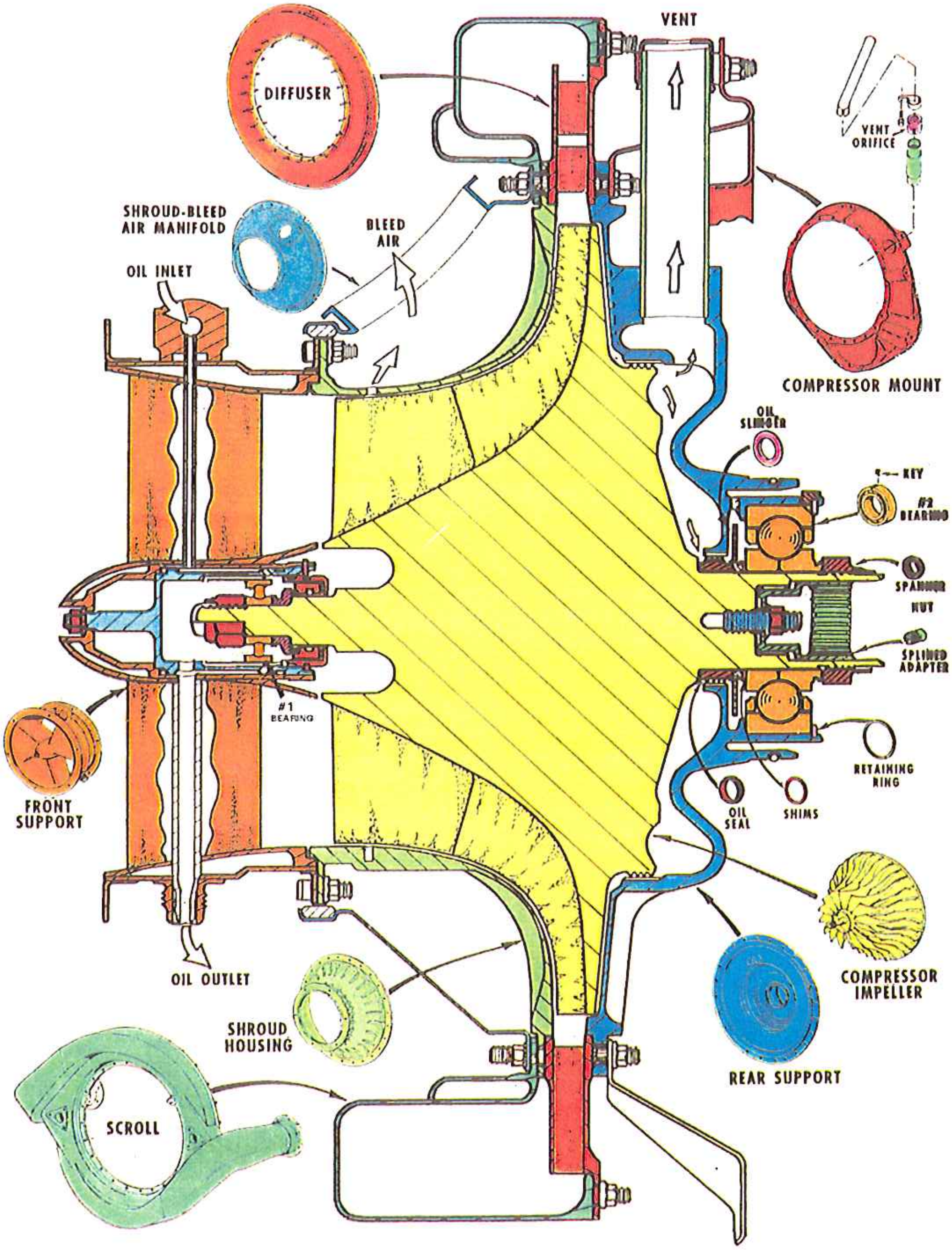

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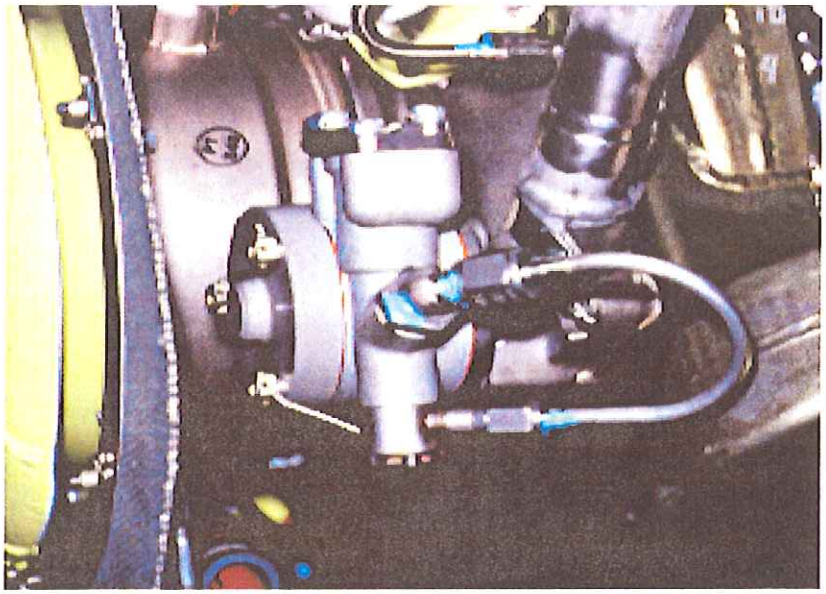
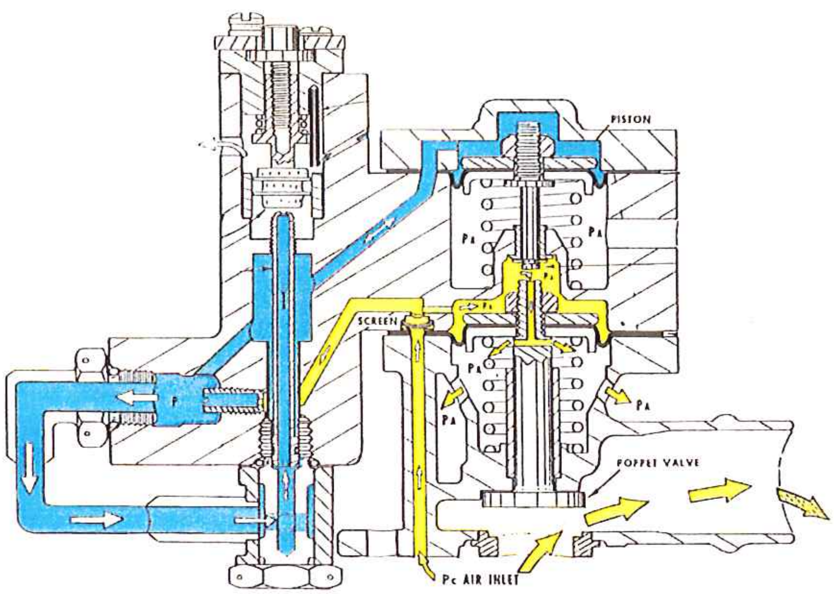
Bleed Valve

The bleed control valve is open during starting and idle operation and remains open until a predetermined pressure ratio is obtained. At this pressure ratio, the valve begins to modulate from open to the closed position. It is normally open during the start cycle and ground idle, modulates during acceleration to full operational speed, and remains closed during flight operation speeds. Pressure sensing for bleed control valve operation is within the valve.





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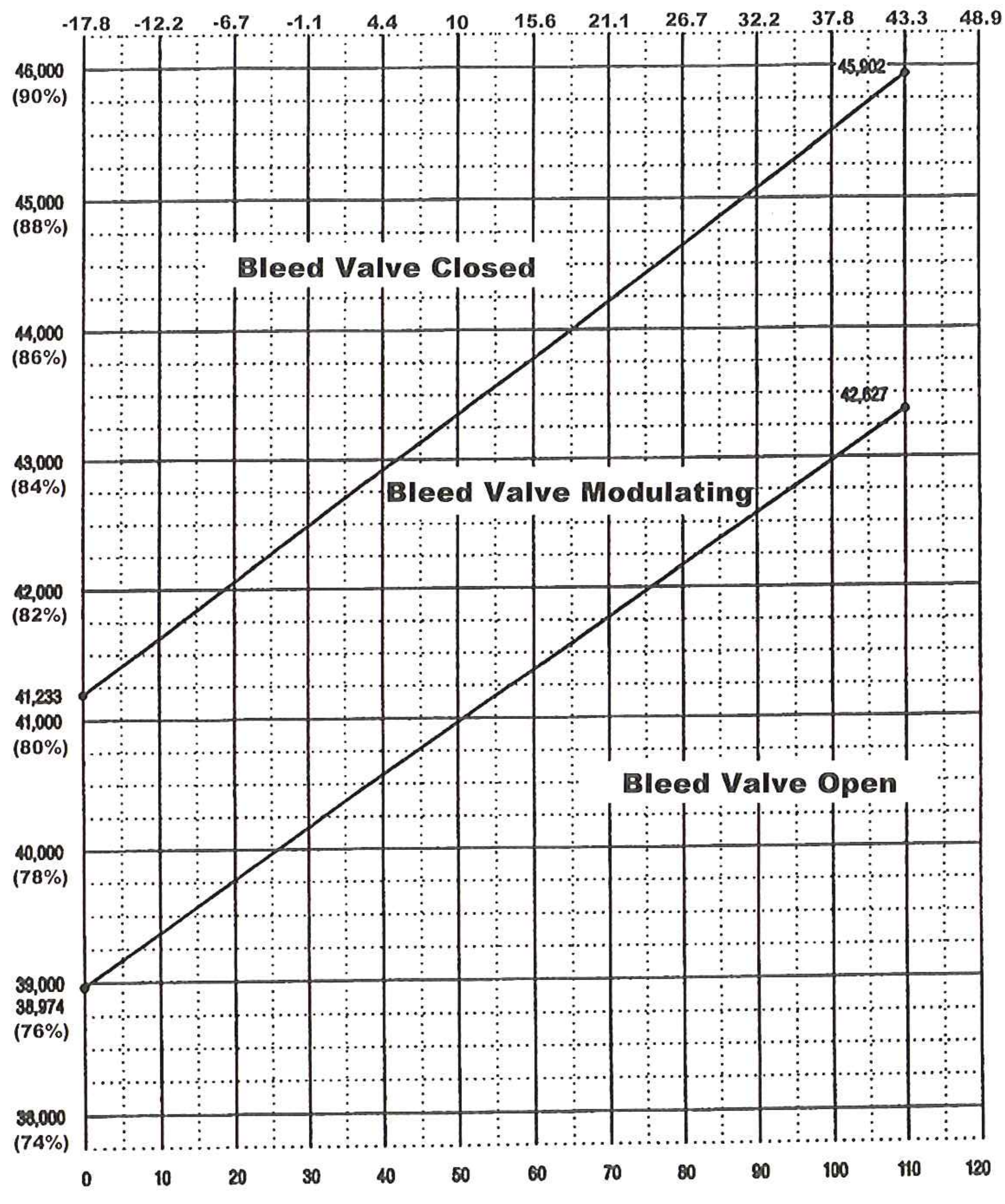
POWERPLANT

This chart shows the relationship between ambient temperature and NG speed, as it applies to the engine bleed valve's modulation from an open to closed position



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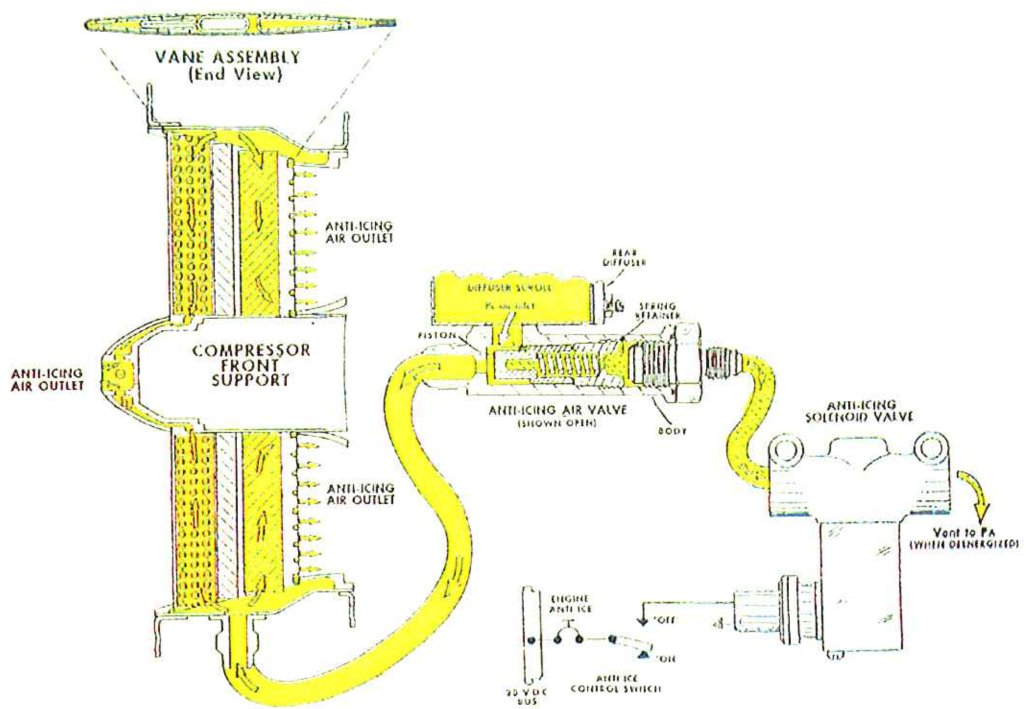
Anti-Icing System

Operation of the engine during icing conditions could result in ice formations on the compressor front support. If ice were allowed to build up, airflow to the engine could be affected and performance decreased. The engine anti-icing system is designed to prevent ice formation on the compressor front support. The pilot activates the system by placing the engine anti-ice switch to the ENG ANTI ICE (on) position. The ENG ANTI ICE white advisory light will illuminate when the system is activated. The ENG ANTI ICE white advisory light is optional on aircraft prior to Serial Number (S/N) 530095 and a standard installation on subsequent aircraft.



When the system is in operation, compressor discharge air, which has been heated due to compression, flows through the anti-icing valve and tube to the compressor front support passages. Hot air flows between the double wall outer shell and into the five hollow radial struts. The hot air flowing through the radial struts exhausts either from small slots in the trailing edge of the struts or from the double wall bullet nose hub of the compressor front support. The compressor inlet guide vanes and front bearing support hub are the only engine components with anti-icing provisions. The bleed air shutoff valve is solenoid controlled. In the event of a total electrical failure, the system is fail-safe to on.

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POWERPLANT

Combustion Section

The combustion section consists of an outer combustion case and a combustion liner. The combustion liner is located inside the outer case. The fuel nozzle is mounted in the aft end of the outer combustion case. The igniter plug is mounted near the aft end of the case. Air enters the single combustion liner at the aft end through holes in the liner dome and skin. The air is mixed with fuel sprayed from the fuel injector and combustion takes place. Combustion gases move forward out of the combustion liner to the first stage gas producer turbine nozzle. Approximately 20% to 25% of the air delivered to this section is required to burn the fuel. The other 75% to 80% is used to cool the internal parts. Approximately 2% of the air is used to seal oil passages.

Air enters the liner at the aft end through holes in the liner dome and skin, where it is mixed with fuel sprayed from the fuel injector and ignited. The combustion gases then move forward out of the combustion liner to the first stage gas producer (NG) turbine section. Most of the cooling air enters the combustion liner in such a manner that the flame pattern is prevented from impinging on the wall of the combustion liner. Also, since the gases of combustion and the cooling air mix before passing through the turbine sections, the resultant temperature is kept within acceptable limits.

Turbine Section

As the gas stream leaves the combustion chamber, it passes to the turbine section. This high-energy gas stream powers the two turbine sections to sustain the airflow through the engine and provide output power. The turbine section consists of a gas producer turbine support, power turbine support, a two-stage gas producer turbine rotor, a two-stage power turbine rotor, and an exhaust collector support. The gas producer turbine drives the compressor and certain engine accessories through the Ng drive train. The power turbine drives the power output shaft, and certain engine accessories.

Measured Gas Temperature (MGT)

The temperature of the gases passing through the turbine is sensed by means of four thermocouples located between the Ng and Np turbine wheels. Each thermocouple senses a voltage proportional to the gas temperature. An average of the four voltages is displayed on the cockpit MGT gage.

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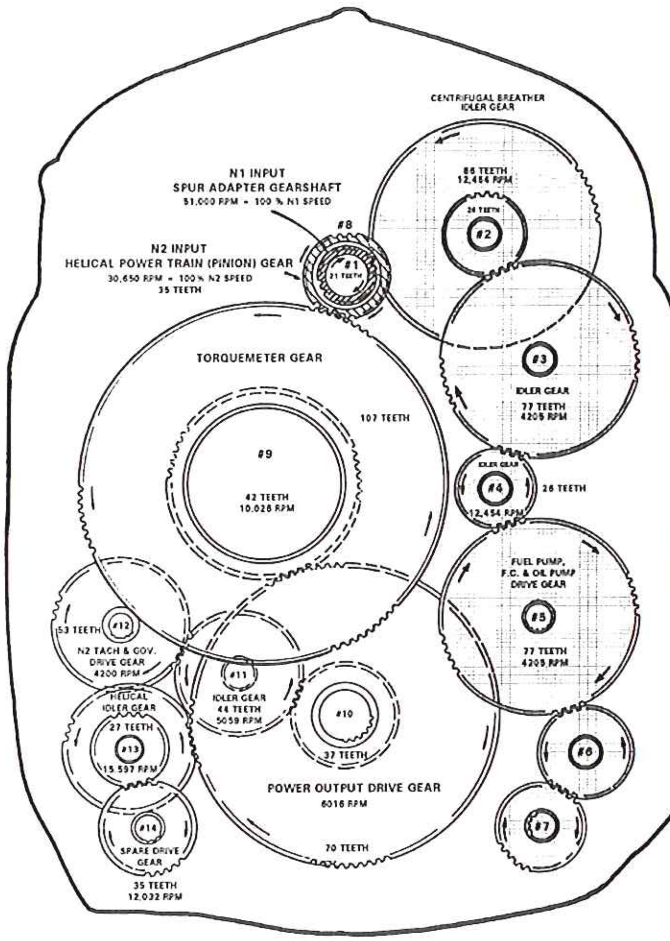
Power and Accessory Gearbox

The main power and accessory drive gear trains are enclosed in a single gear case. The gear case serves as the structural support of the engine. All engine components are attached to the case. There are two independent drive trains in the gearbox, gas producer (Ng), and power turbine (Np).



The gas producer Ng gear train drives the hydro-mechanical unit (HMU), the starter generator, and the oil pump (inside the gearbox).

A two-stage helical and spur gear set is used to reduce Np rotational speed from 32,183 rpm at the power turbine to 6317 rpm at the output drive spline.

The Np gear train drives the permanent magnet alternator (PMA), the torquemeter, and the freewheeling unit.

The gearbox also contains two dual coil monopole sensors. These sensors are magnetic coils that produce an electrical signal. One of the sensors is used for measuring gas producer (Ng) speed. The other is used for measuring power turbine (Np) speed and for overspeed protection. On each sensor, both coils send input signals to the ECU and one signal is also sent to the appropriate indicator; either the Ng gage or the Np needle on the dual tachometer.

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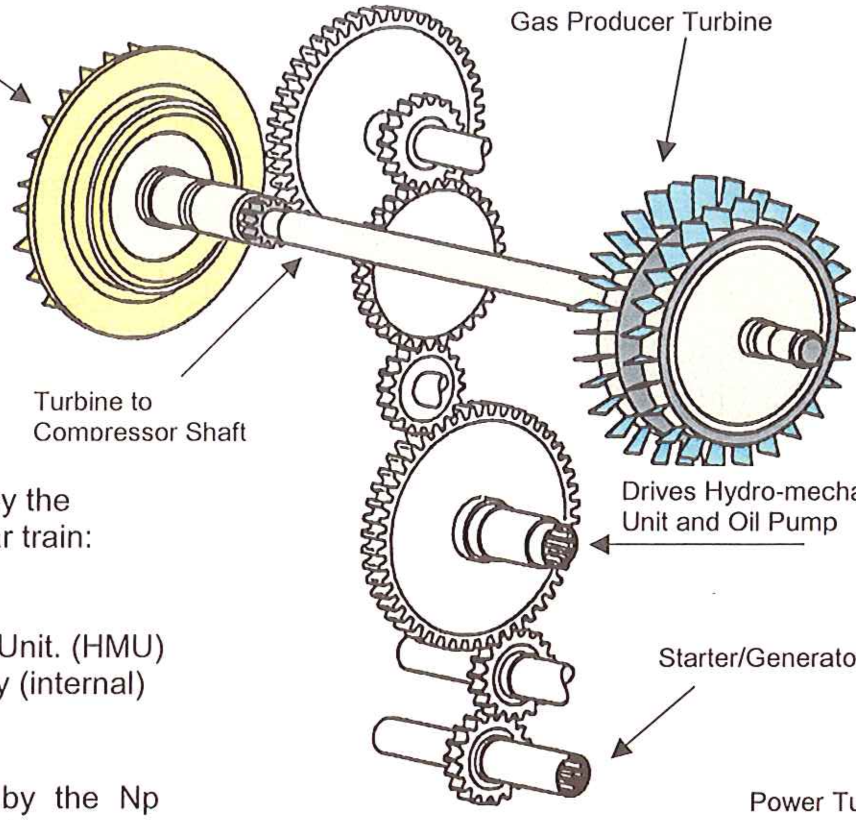
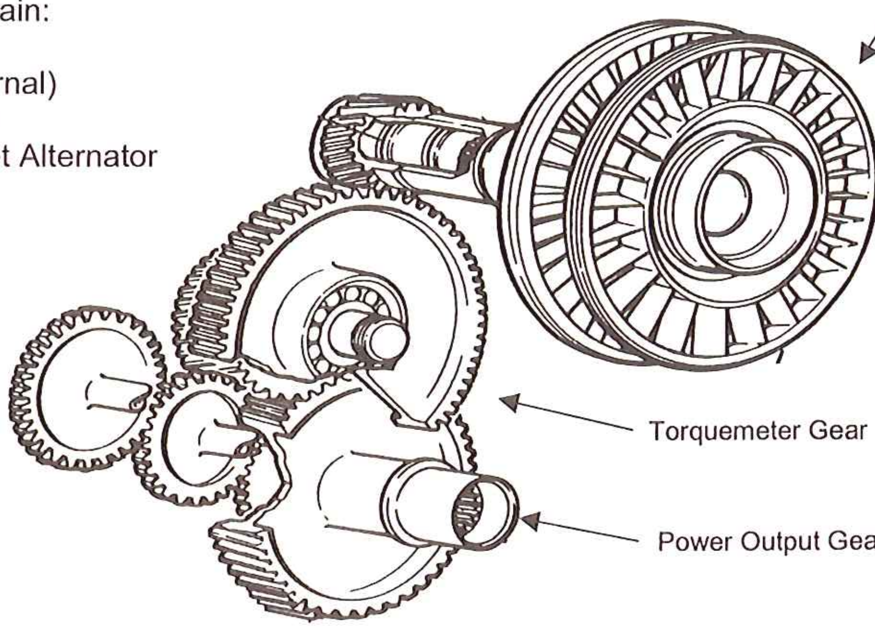


Components driven by the
Ng power turbine gear train:

1. Starter Generator
2. Hydro Mechanical Unit. (HMU)
3. Oil Pump Assembly (internal)

Components driven by the Np power turbine gear train:

1. Torquemeter (internal)
2. Freewheeling unit
3. Permanent Magnet Alternator



Accessory Drive Trains

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Starter/Generator

The starter/generator is used as a DC motor to drive the engine during the starting cycle. Once the engine is started and the generator switch is turned on, it functions as a DC generator to supply power for the electrical needs of the helicopter and to keep the battery charged.

Ignition System

The igniter coil converts 28V DC energy into high temperature/high amperage arcs at the spark igniter gap and is required during the starting cycle. The igniter is threaded into the combustion outer case. It extends into the combustion liner providing ignition sparks that ignite the fuel/air mixture during start. Once the engine is started, the combustion is continuous and the ignition source is no longer required. Electrical power is supplied from the electrical bus and is protected by a 5 amp C/B marked IGNTR.

Fuel System

The engine fuel system consists of the following components: The ECU, CEFA, HMU, fuel nozzle, and burner drain valves.

The electronic control unit (ECU) is part of the FADEC system. The ECU monitors engine power demand and signals the metering valve in the HMU to provide the proper amount of fuel to the combustion chamber.

The combined engine filter assembly (CEFA) filters both fuel and oil supplied to the engine. The fuel portion of the assembly filters the fuel prior to entering the HMU. A bypass valve allows the fuel to bypass the filter in the event it becomes blocked.

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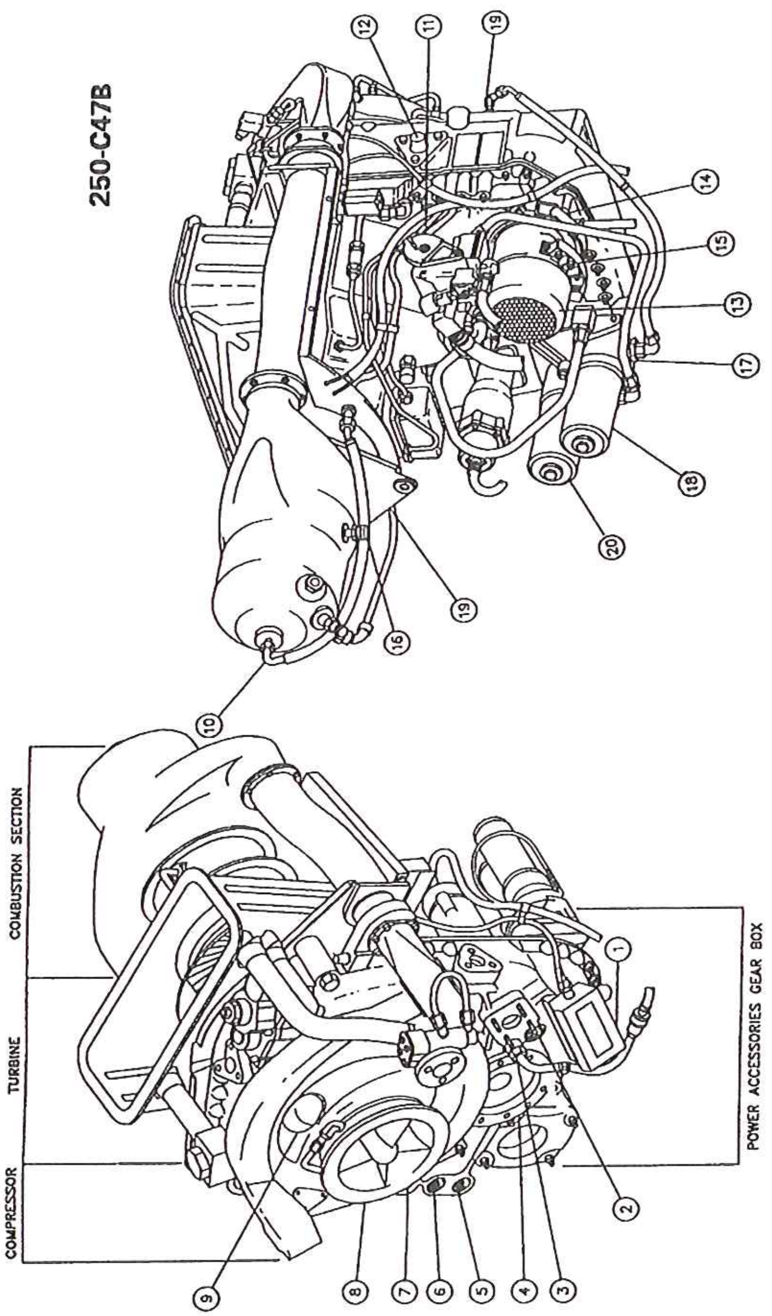

1. Ignition exciter box (ref)
2. Oil tank vent port (ref)
3. Torquemeter oil pressure port (ref)
4. Engine power output pad (ref)
5. Engine oil outlet port (ref)
6. Engine oil inlet port (ref)
7. Engine oil pressure port (ref)
8. Engine intake (ref)
9. Shroud bleed air manifold connection (ref)
10. Fuel nozzle (ref)
11. Fuel control box assembly (ref)
12. Engine mount trunnion (ref)
13. Starter–generator (ref)
14. Starter–generator V–clamp (ref)
15. Starter–generator connections (ref)
16. Hose assembly, combustion chamber drain (ref)
17. Impending fuel filter bypass indicator (ref)
18. Scavenge oil filter (ref)
19. Aft engine mount support (ref)
20. Fuel filter (ref)

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The hydro-mechanical unit (HMU) consists of two parts, a fuel pump and a fuel control assembly. The fuel pump provides high-pressure fuel to the fuel control. The fuel control sends the proper fuel flow to the combustion chamber as directed by the ECU.

The fuel nozzle atomizes and injects fuel into the combustion liner at the proper spray angle to support combustion.

The burner drain valves drain any unburned fuel from the combustion section following an engine shutdown. During start, the drain valves close when the air pressure within the combustion section exceeds the air pressure on the outside of the combustion section by a predetermined value. The valves open on shutdown by means of spring action.

Torquemeter

The torquemeter is a hydraulic type that uses the engine lubrication system as its oil (hydraulic) pressure source. In order to minimize friction effects and provide accurate measurement of torque, the axial gear thrust on the helical torquemeter gearshaft is high. System pressure must always be greater than the torquemeter oil pressure. Therefore, it is necessary to regulate the system oil pressure to the relatively high value of 115-130 PSI.

Torquemeter oil pressure changes with the thrust movement of the helical gear on the shaft. Higher torquemeter oil pressure equates to a higher indicated torque value, and lower pressure corresponds to a lower indicated torque. Torquemeter oil pressure is measured by a transducer and transmitted electrically to the torque indicator.

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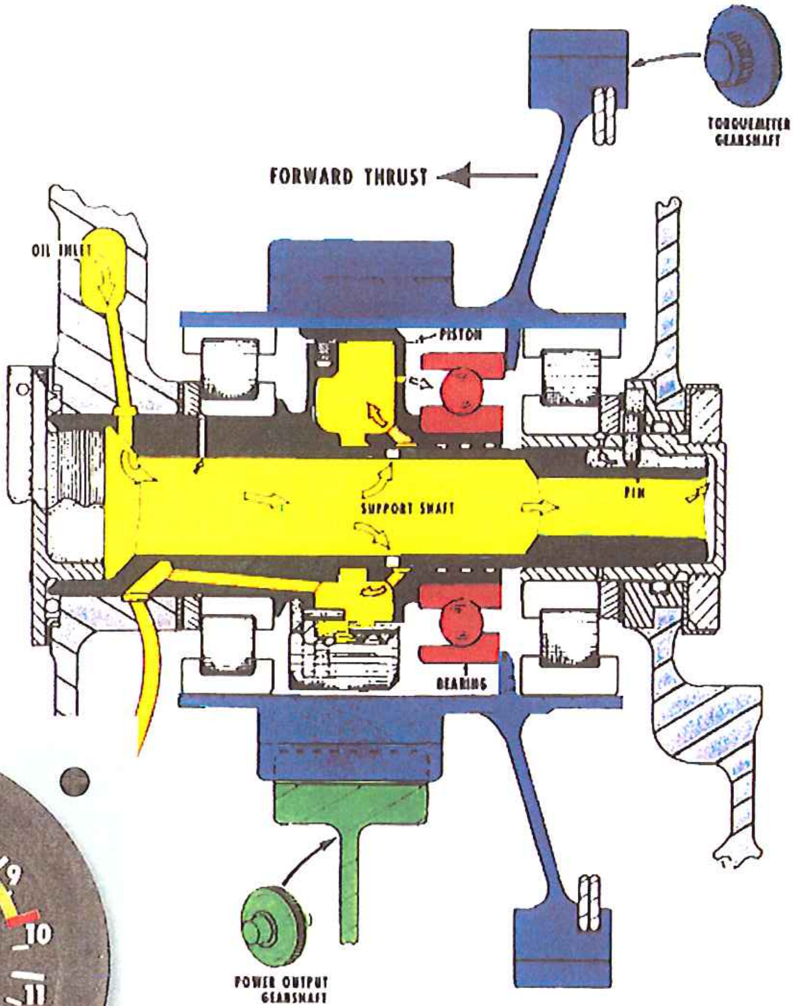
TORQUEMETER SCHEMATIC



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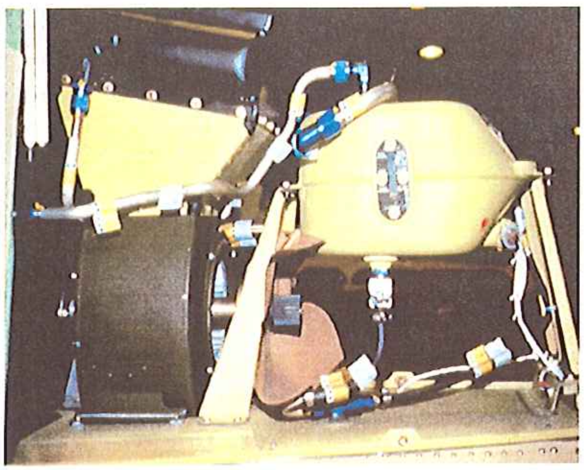
Oil System

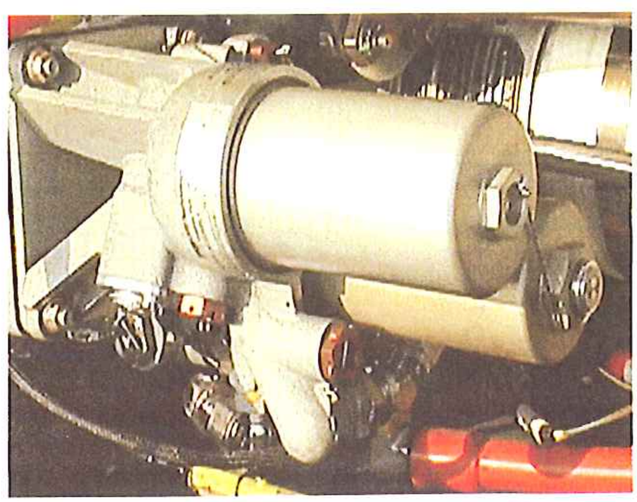
The engine incorporates a dry sump oil system with an externally mounted supply tank and an oil cooler located on the top aft section of the fuselage. Oil is supplied from the tank to pressure and scavenge pumps mounted within the engine accessory drive gearbox. The oil filter assembly, consisting of an oil filter, filter bypass valve, and pressure regulating valve, is located externally in the upper left-hand side of the gearbox. Magnetic chip detectors are installed at the bottom of the gearbox and at the engine oil outlet connection. All engine oil system lines and connections are internal except the pressure and scavenge lines to the front compressor bearing and to the bearings in the gas producer and power turbine supports.



The system is designed to furnish adequate lubrication, scavenging, and cooling as needed to the bearings, splines, and gears, regardless of the helicopter attitude or altitude. Jet lubrication is provided to all compressor, gas producer turbine, and power turbine rotor bearings, and to the bearings and gear meshes of the power turbine gear train, with the exception of the power output shaft bearings. The power output shaft bearings and all other gears and bearings are lubricated by oil mist.

The oil pump and filter assembly supplies a pressurized volume of oil for proper lubrication and cooling of the bearings and gears. The pump also has the capacity to scavenge oil from the various sump cavities and to return this oil to the supply tank.

The Combined Engine Filter Assembly (CEFA) provides both fuel and scavenge oil filtration within a single filter assembly. The assembly consists of a fuel filter bowl, fuel bypass valve, fuel differential pressure indicator, manifold assembly, a disposable fuel filter element, an oil filter bowl, oil bypass valve, oil differential pressure indicator, and a disposable tube filter element. The CEFA is located on the lower left hand portion of the power and accessory gearbox assembly.



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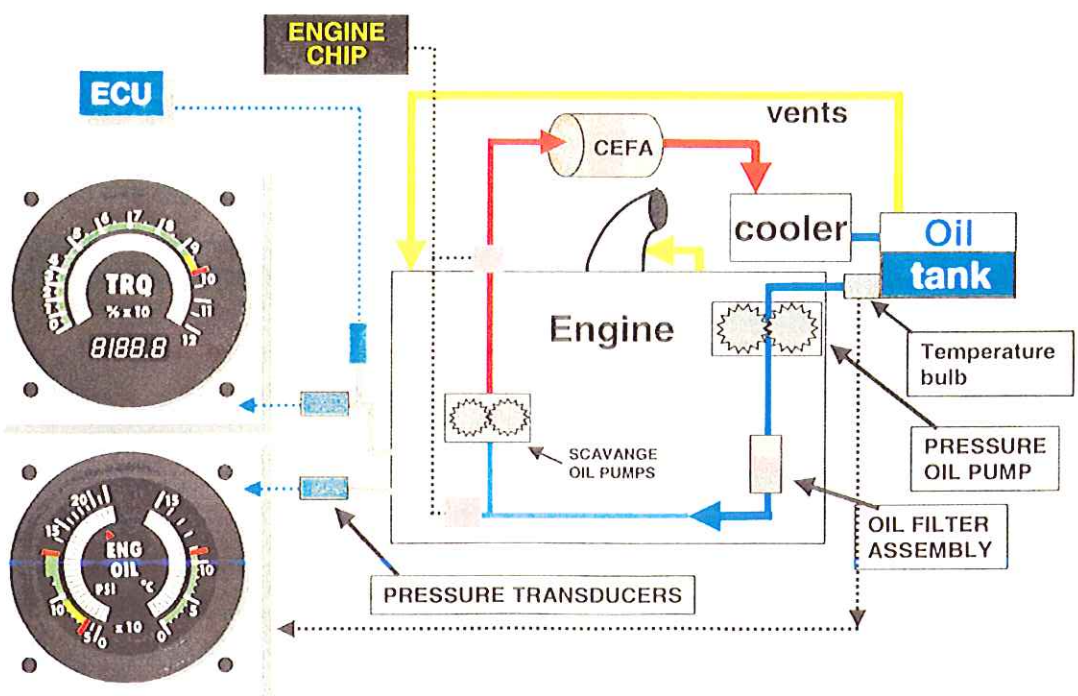
The bypass valves for both the fuel and scavenge lube filters allow flow to go into a bypass condition if excessive differential pressure occurs across the filter elements. The differential pressure indicators provide a visual signal when differential pressure across either element exceeds a predetermined value indicating that the element is contaminated and requires replacement. Both indicators are reset manually following replacement.

Oil from the tank is sensed for temperature, delivered to the accessory gearbox, sent to the pressure pump, then to the filter, and then to various points of the engine for lubrication and cooling. The engine oil that circulates through the torque meter flows over the lower engine magnetic chip detector. All the oil is then picked up by the four scavenge oil pumps, flows over a second magnetic chip detector, and proceeds to the CEFA for filtration. Return oil is routed from the engine oil outlet port to the oil cooler. A cooler blower assembly is mounted on the tail rotor drive shaft and provides cooling air to the oil cooler. The oil passes through the cooler element, and then returns to the tank.



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FADEC

The 407 is, in many areas, a totally new design. This is particularly true in the area of the Full Authority Digital Electronic Control (FADEC) system. The FADEC includes numerous features and benefits never before available in a light, single-engine Bell helicopter. The FADEC system is designed to enhance flight safety and reduce pilot workload as well as provide other important benefits. In addition to the operational benefit of increased TBO, engine automatic start, and precise control of main rotor speed, are features such as redundant signal sensing, continuous monitoring, and self-diagnostics. Since much of the redundant design is transparent to the pilot, the caution/warning/advisory system has been expanded to advise of conditions resulting from the increased monitoring.

Although the possibility of a FADEC system failure is unlikely, Pilots and Maintenance Personnel must have an operational understanding of the FADEC system, along with a sound knowledge of emergency and troubleshooting procedures. Bell Helicopter recommends that personnel involved with the 407 familiarize themselves with the procedure for FADEC FAILURE, as prescribed in the Emergency Procedures Section of the Rotorcraft Flight Manual.

The FADEC uses a single channel control with 1 microprocessor and 1 electronic lane. There is also a MANUAL mode hydro mechanical backup. The FADEC system has two main components. The airframe mounted Electronic Control Unit (ECU) and the engine mounted Hydro Mechanical Unit (HMU).

The ECU monitors numerous internal and external inputs to modulate fuel flow and control engine speed, acceleration rate, temperature, and other engine parameters. The ECU provides inputs to the HMU to modulate fuel flow based on the continuous monitoring of the following: Measured Gas Temperature (MGT), Gas Producer speed (Ng), Power Turbine speed (Np), Main Rotor speed (Nr), Engine Torque Meter Oil Pressure (TMOP), Collective Pitch (CP) and rate, Compressor Inlet Temperature (CIT), Ambient Pressure (P1), and Power Lever Angle (PLA)/throttle position. During flight in AUTO mode with the throttle in FLY detent position (PLA 70°), the FADEC has complete control over engine operation to maintain NR within limits.

The HMU consists of a two stage suction fuel pump, fuel metering assembly, Auto/Manual changeover solenoid valve, electric overspeed valve, mechanical fuel shut off valve, hot start fuel solenoid valve, and altitude compensated bellows. The HMU provides fuel modulation via a stepper motor in AUTO mode and a Hydro Mechanical actuator in MANUAL mode.

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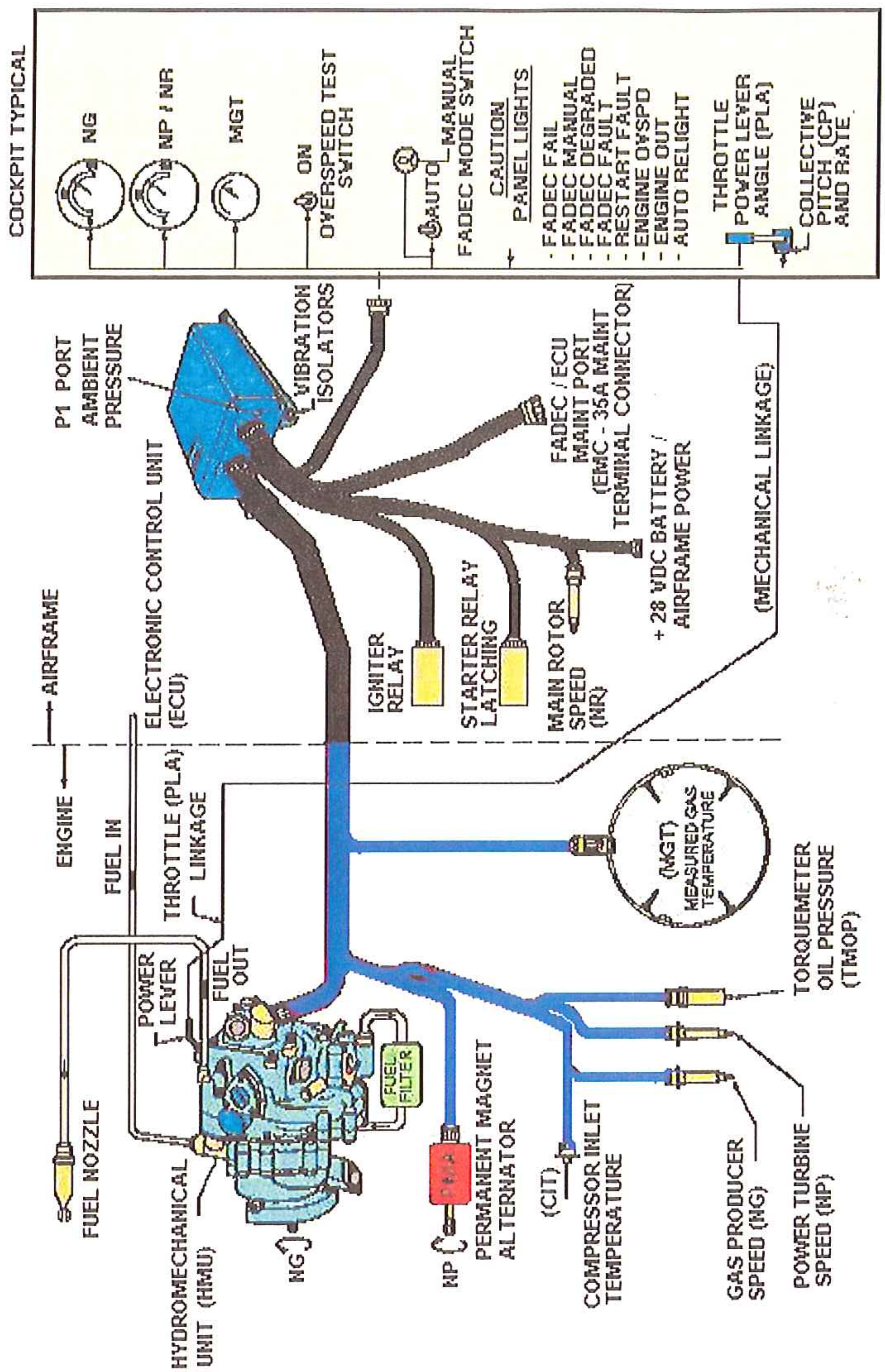

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Power Up Mode And Built In Test (Bit)

The FADEC system incorporates logic and circuitry to perform self-diagnostics. In general, sensors are checked for continuity, rate, and proper range. Discrete inputs are checked for continuity and output drivers are monitored for current demand to sense failed actuators and open or shorted circuits. A FADEC power up check exercises output drivers and actuators to ensure system functionality and readiness. If any faults are detected during the self-test, the appropriate FADEC caution panel light will illuminate and alert the pilot.

The helicopter 28vdc bus supplies electrical power to the FADEC ECU until the engine achieves 85% Np. Above this speed, the engine power turbine driven Permanent Magnet Alternator (PMA) is available as a dedicated power source for the FADEC ECU. The FADEC ECU will select the higher voltage source between 28VDC bus and the PMA.

Start In Auto Mode

An automatic start function is designed into the 407. Throttle modulation of fuel flow by the pilot is not required because fuel flow is automatically controlled by the FADEC. "Hot Start Abort Logic" is incorporated into the system which is designed to cut of fuel flow and abort a start if start MGT exceeds 843ºC, at pressure altitudes less than 10,000 feet and if residual MGT was less than 82.2ºC at initiation (5% Ng) of start, start MGT exceeds 912ºC at pressure altitudes greater than 10,000 feet or if residual MGT was greater than 82.2ºC at initiation (5% Ng) of start, or voltage to the FADEC ECU drops below 10.3 vdc.

Although the start sequence is automatic, the pilot is responsible for monitoring the start process and taking appropriate action if required. Therefore, it is recommended that both the throttle and start switch are guarded until the start is completed. Do not initiate a start if FADEC related caution panel lights are illuminated unless appropriate maintenance corrective action has been performed.

Start In Manual Mode

In accordance with the Rolls-Royce Allison 250-C47B Operation and Maintenance Manual, Manual Mode starting on the ground is not authorized except for use under emergency conditions or under special permit from the local aviation authority. Refer to the Rolls-Royce Allison 250-C47B Operation and Maintenance Manual to ensure all Manual Mode Operational Procedures are followed. Automatic hot start abort features are not available in MANUAL mode.

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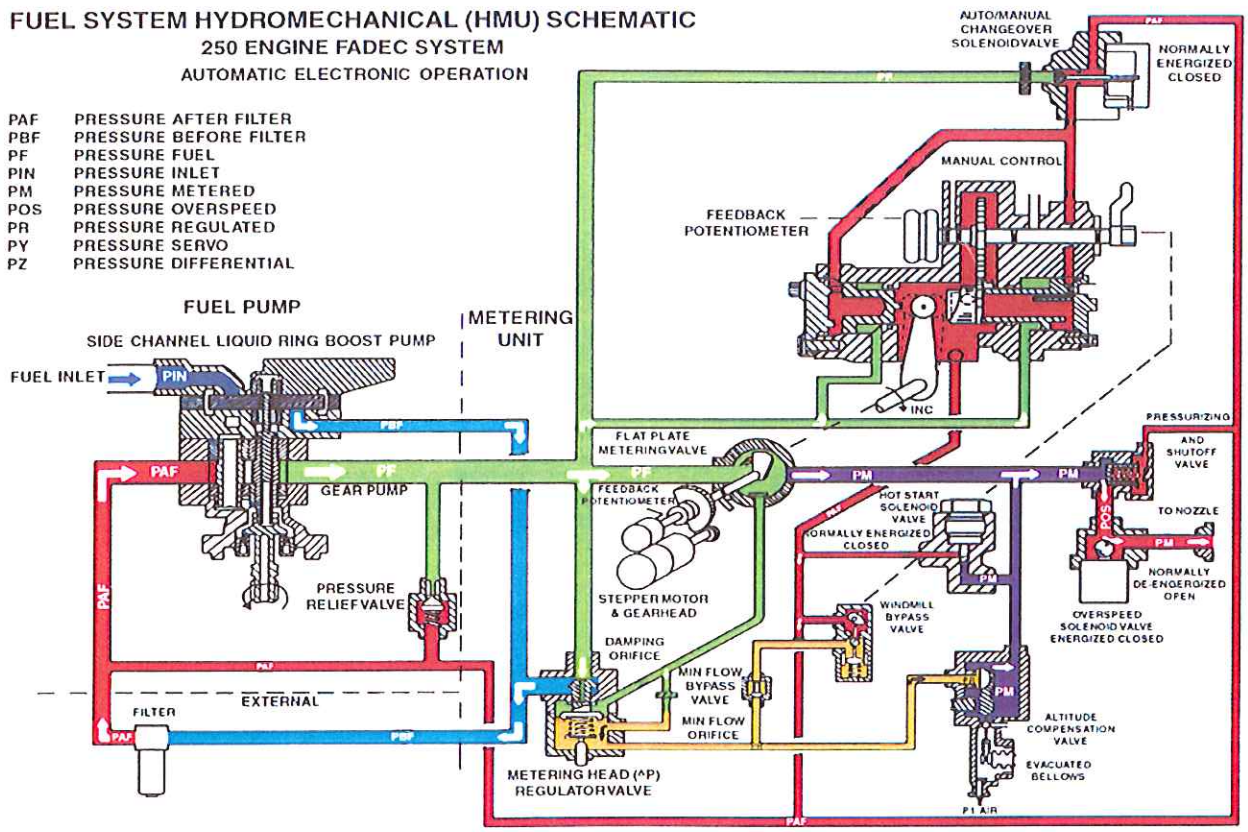
Inflight Operation In Auto Mode

During flight in AUTO mode with the throttle in FLY detent position (PLA 70º), the FADEC has complete control over engine operation to maintain Nr within limits.



The ECU receives engine and airframe parameter inputs and cockpit command signals, processes them, and modulates the HMU Stepper Motor to provide adjustment to the Fuel Metering Valve and achieve desired engine performance. Precise electronic control provides automatic maintenance of rotor RPM throughout the normal range of operation and lessens pilot workload.

In AUTO mode, the FADEC is capable of detecting an engine flameout by sensing Ng deceleration. Without any pilot action, the auto-relight sequence is initiated by establishing a controlled fuel flow and activating the ignition system. The FADEC will control the MGT and accelerate the engine to the proper state.

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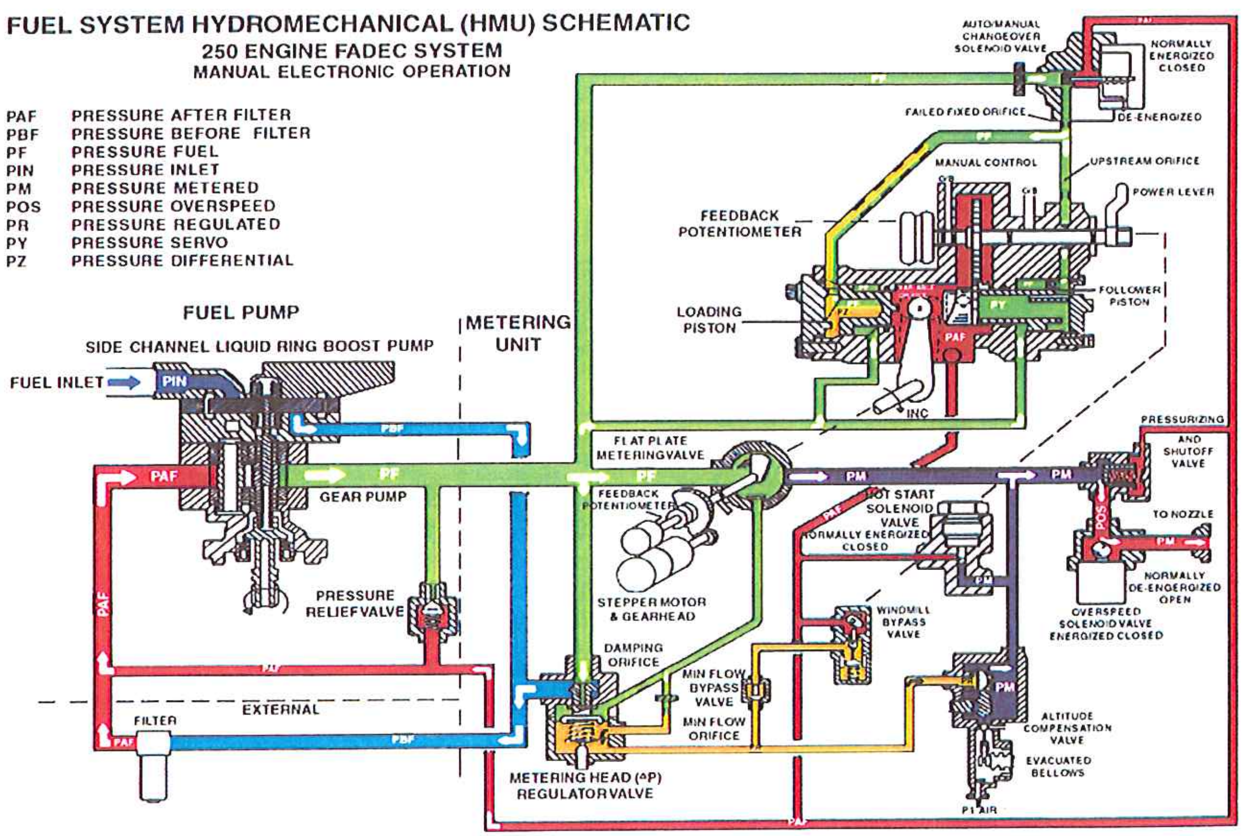
Inflight Operation In Manual Mode

Whereas, in AUTO mode, the stepper motor receives electronic signals from the FADEC ECU and adjusts the fuel metering valve to maintain proper engine performance; in MANUAL mode, the fuel metering valve responds to the position of the throttle and the PLA. In MANUAL mode, high pressure fuel is allowed to enter both sides of the HMU interior and move the pistons from their parked positions into contact with the fuel metering valve lever. As PLA position is changed by throttle manipulation, variable fuel pressure is achieved in the follower piston side of the HMU. By varying fuel pressure in the follower piston side of the HMU through throttle manipulation, movement of the fuel metering valve lever (and hence the fuel metering valve) is achieved, allowing the pilot to effectively control engine RPM.



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Fadec System Faults

The FADEC ECU continuously monitors the FADEC system for faults and makes appropriate accommodations to continue operations. Eight lights in the caution/warning/advisory panel are controlled by the FADEC. If any failure occurs in the ECU/HMU or in one of the input/output signals that significantly impacts the ECU or control of the HMU and requires pilot action; the pilot will be alerted by the FADEC FAIL warning horn and the FADEC FAIL/FADEC MANUAL warning lights. If the detected failure does not significantly impair the functioning of the ECU, the pilot will be alerted by a FADEC DEGRADED caution light, FADEC FAULT caution light, RESTART FAULT advisory light, or a combination thereof, depending upon the nature of the fault.

If a fault is minor in nature, it will not be communicated to the pilot with the engine running. These faults are identified as maintenance advisory faults and will be displayed during shutdown via a FADEC DEGRADED light when the throttle is placed in the cutoff position and NG speed decays below 9.5%.

Additional detailed information regarding FADEC operation is located in the Manufacturer's Data Section of the Rotorcraft Flight Manual. Pilots and Maintenance Personnel are encouraged to research this section for a more thorough understanding of FADEC operation and characteristics.

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Servicing

Certain oils conforming to following specifications are approved for use in engine:

Specification OAT Range

M1L-L.7808 (NATO 0-148) Any OAT

MIL-L-23699 (NATO 0-156) OAT above -4O°C(-40°F)

DOD-L-85734 OAT above -40°C (—4O°F)

NOTE

As per Allison the preferred engine oils for MIL-L-23699 are Mobile Jet Oil 254 and Aeroshell 560.

NOTE

Because of availability, reduced coking and better lubricating qualities at higher temperatures, qualified MIL-L-23699 oils are preferred by engine manufacturer.

NOTE

Long term use of 000-L-85734 oil may increase probability of seal leakage in accessory gearbox.

Refer to BHT-407-FM-1 for engine oil limitations.

Capacity: 6.0 U.S. quarts (5.7 liters).

Engine oil tank is located under aft fairing, and access doors are provided for filling and draining oil tank. A sight glass and filler cap dipstick are provided to determine quantity of oil in tank.

NOTE

If helicopter engine has been shut down for more than 15 minutes, scavenge oil could have drained into gearbox. Dry motor run engine for 30 seconds before checking oil level. It not accomplished, a false high engine oil consumption rate indication or overfilling of oil tank could result.

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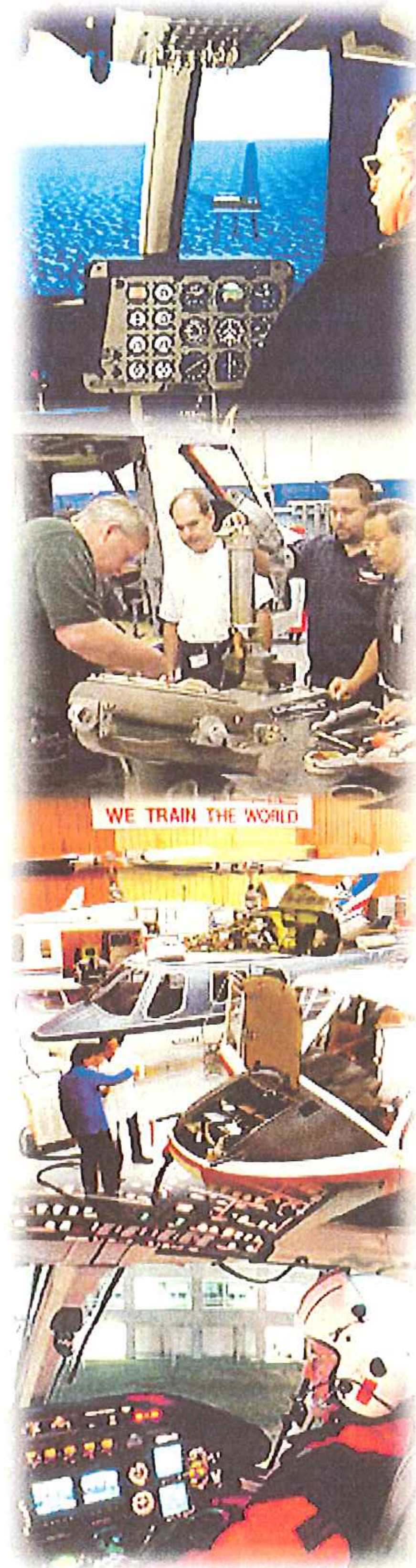

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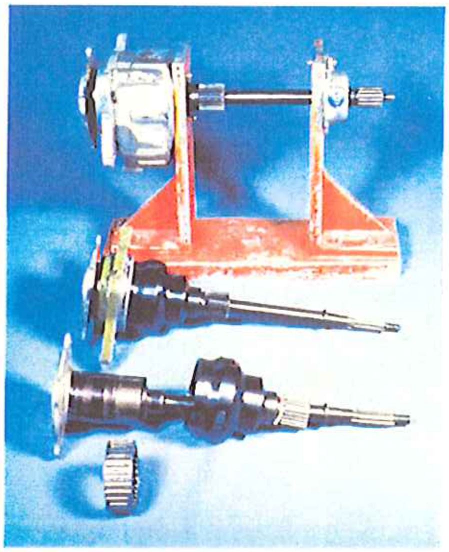
DRIVETRAIN AND ROTORS

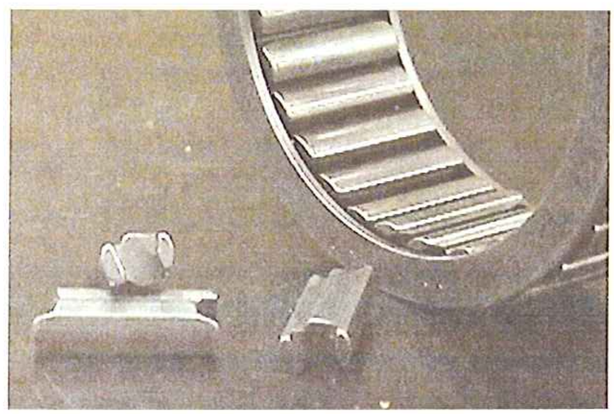
The drivetrain system provides a means of transmitting power from the engine to the main and tail rotor assemblies. The drivetrain includes the freewheeling unit assembly, main drive shaft, transmission, mast, tail rotor drive shaft, oil cooler blower, and tail rotor gearbox. The rotors are the main rotor and the tail rotor.

Freewheeling Unit Assembly



The freewheeling unit is mounted on the engine gearbox and driven under power from the engine power takeoff gear shaft. Engine power is transmitted to the outer race of the freewheeling unit, through the engaged sprag clutch and into the freewheeling inner shaft. The freewheeling inner shaft couples the engine to the main driveshaft on the forward attachment plate. The tail rotor drive system is driven through a flexible coupling and a splined adapter mounted on the aft end of the freewheeling inner shaft.

During autorotation, the sprag clutch disengages and the rotational forces of the main rotor are utilized to drive the transmission accessories and tail rotor drive system.



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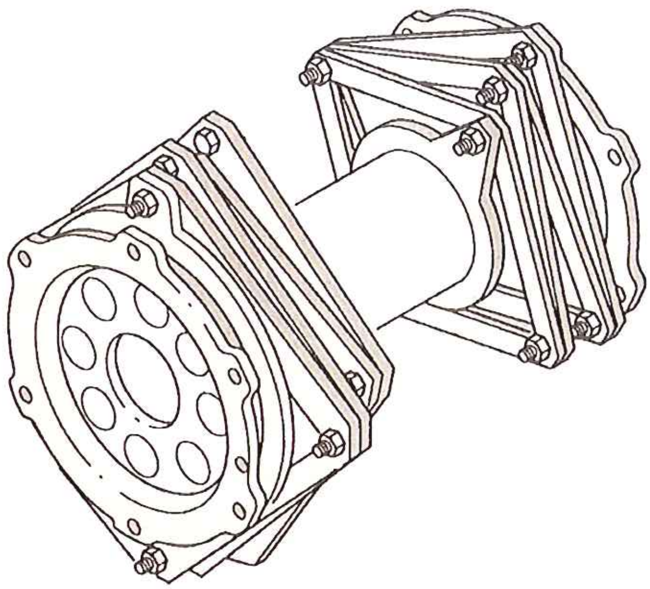
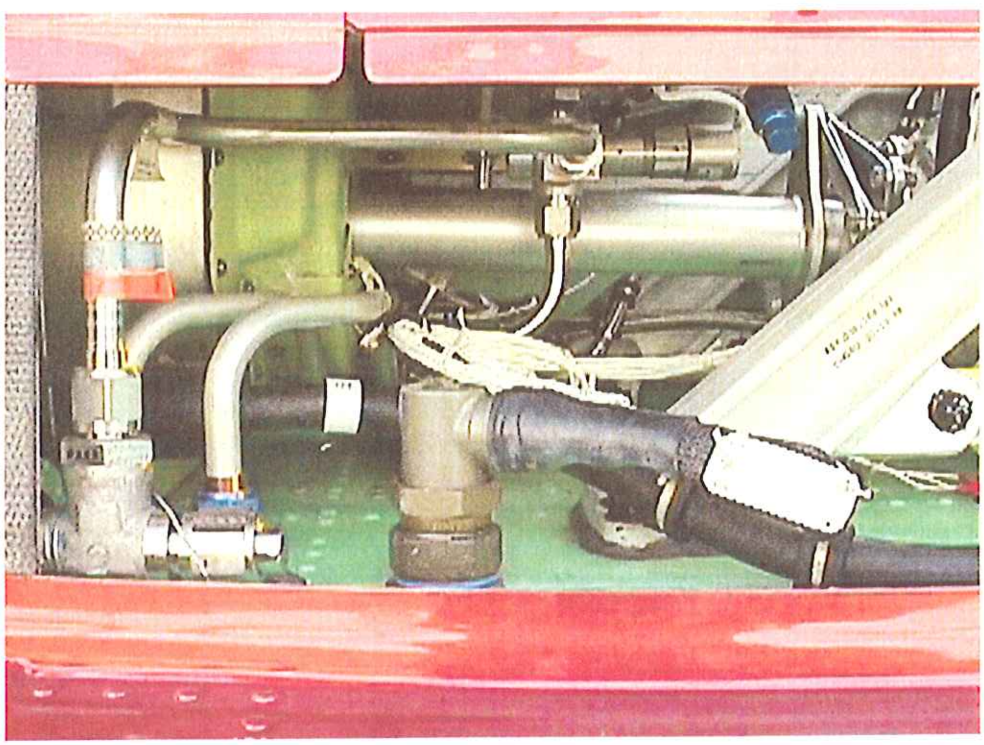
DRIVETRAIN AND ROTORS

Main Driveshaft



The engine to transmission Kaflex main driveshaft provides a flexible drive connection between the freewheeling unit and transmission. This flexible shaft allows for the smooth transfer of torque when the transmission and freewheeling unit are not aligned. Flexibility is provided by three rectangular plate sets in each coupling. Each plate flexes, providing both angular misalignment and length changes to accommodate movement of the transmission on its mounts. Each coupling can be considered a truss work in which torque loads are carried as axial loads in straight members of each plate. The main driveshaft turns 6317 RPM at 100 % Np.



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DRIVETRAIN AND ROTORS

Transmission

The transmission is utilized to transfer engine torque through the mast assembly to the main rotor system. The two stage transmission will reduce input RPM from engine (6317) to 413 rotor RPM, a reduction ratio of 15.29:1.

The transmission assembly consists of a top support case and a lower case. This assembly contains the input pinion, bevel gear arrangement, planetary gear train, and accessory gear drive. Components attached to the transmission and mast assembly are the main driveshaft, transmission oil pump, transmission oil filter housing, hydraulic pump, rotor rpm monopole pickup, and two magnetic chip detectors.

The transmission assembly is attached to the helicopter roof by a pylon installation. The pylon installation uses two side beams, four elastomeric corner mounts, and two fore/aft restraint springs.

Elastomeric bearings are used to minimize vibrations being transferred into the airframe. The transmission is equipped with chip detectors and a temperature switch to provide an indication of a related system malfunction.

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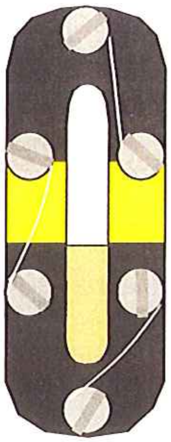
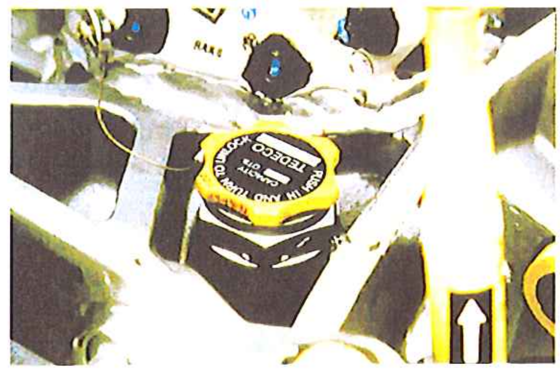
Transmission Oil System

The transmission oil system provides lubrication to the transmission, mast and freewheeling unit assembly.



An oil level sight glass is located on the right side of the transmission lower case and may be viewed through a cutout in the air induction cowling. A non-vented filler cap is located on the right side of the top case to service the transmission oil system.



The transmission accessory gear drives the transmission oil pump. The pump output is between 6.0 to 6.7 gallons per minute. The pressure is regulated by the transmission oil pressure regulator valve and is set at approximately 52 PSI. The pump scavenges oil from the lower case sump through a wire screen and the lower chip detector. Oil is then directed to the transmission mounted oil manifold and the filter element.

To ensure oil flow is not restricted, the filter incorporates an impending bypass indicator button on the end of the filter housing which will extend at 14± 2 PSID. The filter bypass valve will open and the oil will totally bypass the filter element at approximately 17 PSID.

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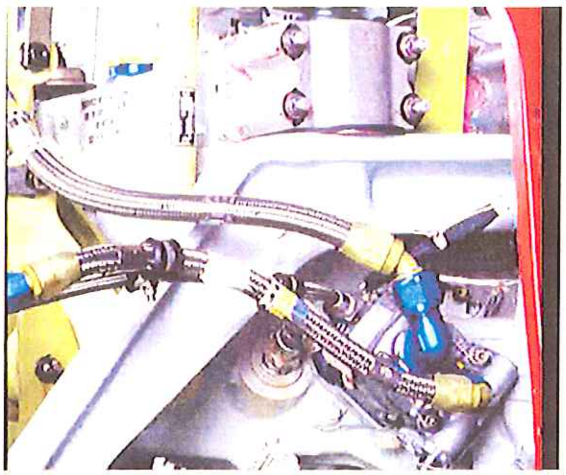
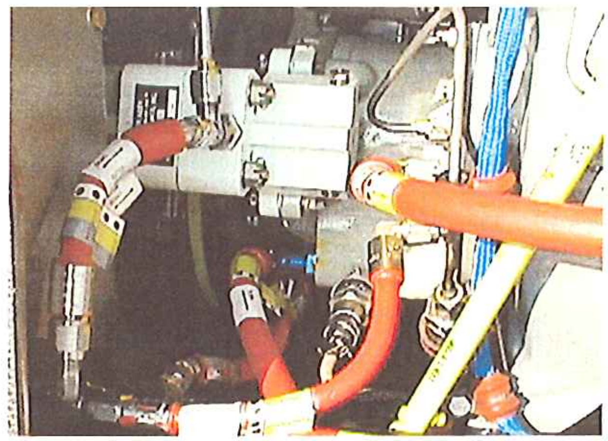
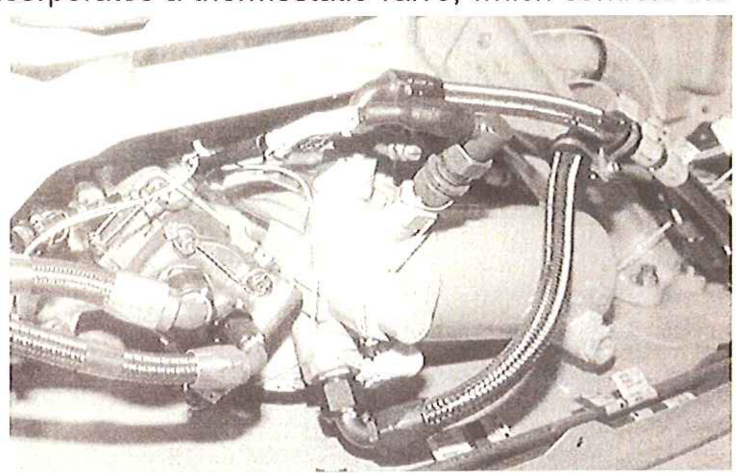
DRIVETRAIN AND ROTORS

Transmission oil is monitored for metallic contamination by three magnetic chip detectors. Two of these chip detectors are located on the left side of the transmission case, one upper and one lower. The third chip detector is found on the freewheeling unit. The chip detector consists of a self-locking bayonet probe with a permanent magnet at the end. Free ferrous metal particles in the oil are attracted to the magnet. When sufficient metal is attracted to complete the circuit between pole and ground, the TRANS CHIP detector segment on the caution panel will illuminate.





The transmission oil filter manifold incorporates a thermostatic valve, which controls the flow of oil to the oil cooler. At oil temperature below 150°F (66°C), the oil cooler is bypassed and oil is returned into the transmission. As the oil temperature increases above 150°F (66°C), oil is gradually directed to the oil cooler until 178°F (81°C), all oil is directed through the cooler. A temperature bulb for the oil temperature indicator and a thermoswitch for the XMSN OIL TEMP light are also located on the transmission oil filter manifold.



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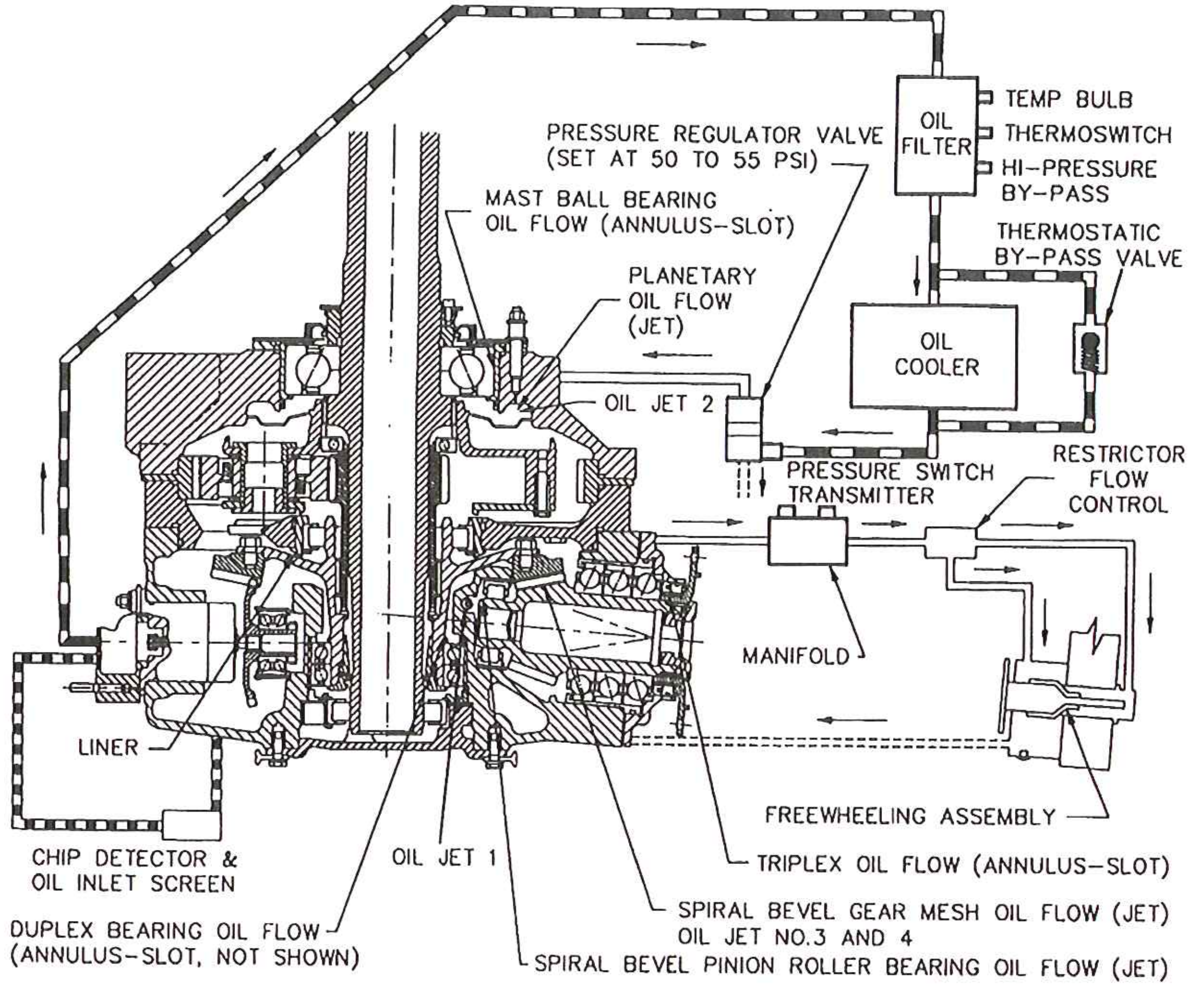

NOTE:

PUMP: Design flow rate = 6 to 6.7 gpm
Rated pressure = 80 psi

LEGEND

Supply oil at pump pressure, 80 to 120 psi.
Supply oil at regulated delivery pressure, 50 to 55 psi.
Return oil at atmospheric pressure.
Scanvenge oil

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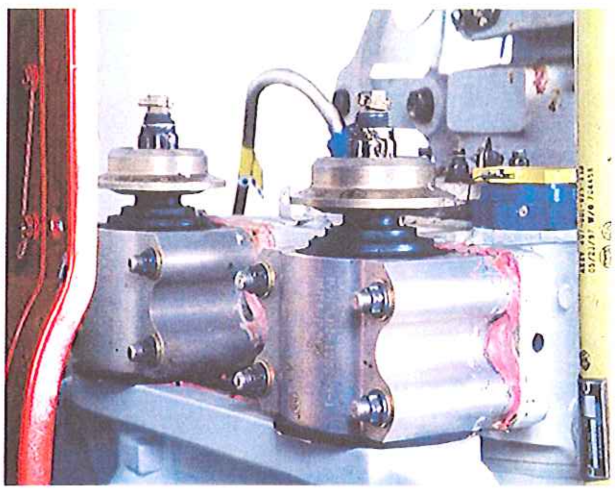
DRIVETRAIN AND ROTORS

After the oil exits the cooler (or bypasses the cooler through the thermostatic by-pass valve), it is directed to the transmission to lubricate the various gears and bearings. The oil is then directed to the oil manifold located below the transmission input driveshaft. A pressure transducer and oil pressure switch are mounted on the transmission deck oil manifold. The transducer provides signals to the oil pressure gage and the pressure switch controls the illumination of the XMSN OIL PRESS light.

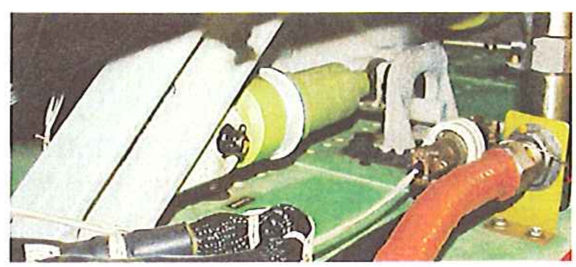
After leaving the oil manifold, oil flows through the forward firewall and into a "T" fitting. The "T" fitting is equipped with two restrictors. The restrictors reduce flow and direct oil into the freewheel forward duplex bearing and aft housing bearing. After lubricating the bearing in the aft housing, oil moves forward through the hollow engine output driveshaft to the freewheel sprag clutch and bearing where it is collected in the forward freewheel housing and returned to the main transmission.

Transmission Mounts



The transmission is mounted to the helicopter roof by two pylon assemblies. The pylons are bolted to the aircraft roof in such a manner to give the transmission and mast a one degree left tilt to compensate for translating tendency and a five degree forward tilt to maintain level cabin attitude in forward flight. Four elastomeric bearings (corner mounts) bolt to the transmission housing and to the pylons. These bearing assemblies are the primary vibration isolating components that provide for a vibration free ride throughout the helicopter's speed range.

At the base and rear of each pylon are two elastomeric bearing transmission restraints. These restraints are used to maintain the proper alignment of the engine to transmission. On the forward end of each restraint are Stop Deck Fittings that limit transmission movement.



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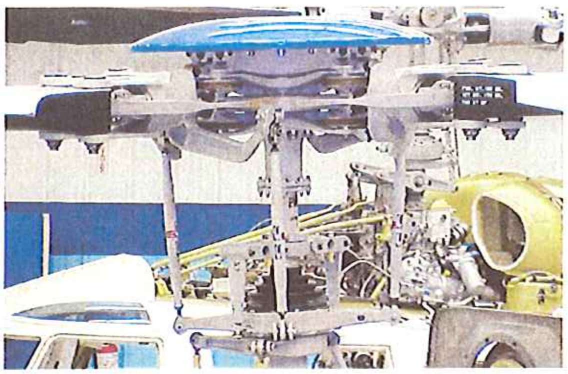
DRIVETRAIN AND ROTORS

Mast

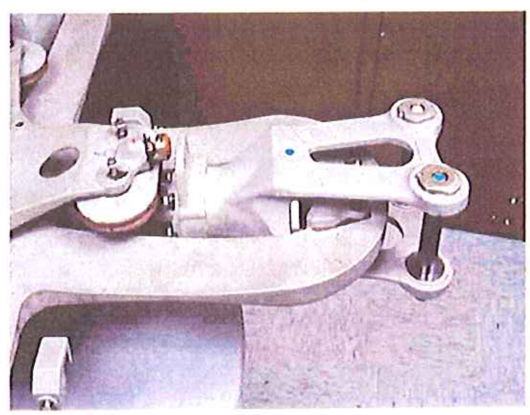
The mast is attached to the transmission by means of a mast locking plate and studs in the top of the transmission case. The mast is a hollow steel shaft that transmits rotational energy from the transmission to the rotor system. It incorporates three sets of splines and two threaded areas into its design. The upper spline is utilized to position and attach the main rotor. The swashplate drive splines are designed to receive a collar set that connects and provides drive to the swashplate drive links. The lower splines are the drive splines where the rotational energy from the transmission is transmitted to the mast from the planetary section. The upper portion of the mast is threaded to receive the main rotor retaining nut while the lower threads are used to secure the mast bearing retaining nut.



The main rotor is a 35 foot diameter, soft - in - plane flex beam (flapping flexure) type yoke/hub with four interchangeable blades. Elastomeric technology is incorporated and allows for blade movement. The blades and yoke are all composite. The rotor is designed to rotate at 413 rpm at 100% Nr.

Main Rotor Hub

The main rotor hub contains a glass/epoxy composite yoke that acts as a flapping flexure. A flapping stop and a droop stop located at the inboard portion of each spindle protect and limit the composite yoke from excessive flexing. Cyclic centering is incorporated into this aircraft to reduce stress on the yoke and mast when the aircraft is on the ground (weight on gear). The feathering bearings and lead lag dampers are elastomeric elements that require no scheduled maintenance and have benign failure modes. Blade flapping is accomplished by a virtual hinge in the yoke (flapping flexure). The hub plate, pitch horns, and spindles are made of aluminum forgings for strength and reduced weight.



The feathering bearings and lead lag dampers are elastomeric in design and require no scheduled maintenance.

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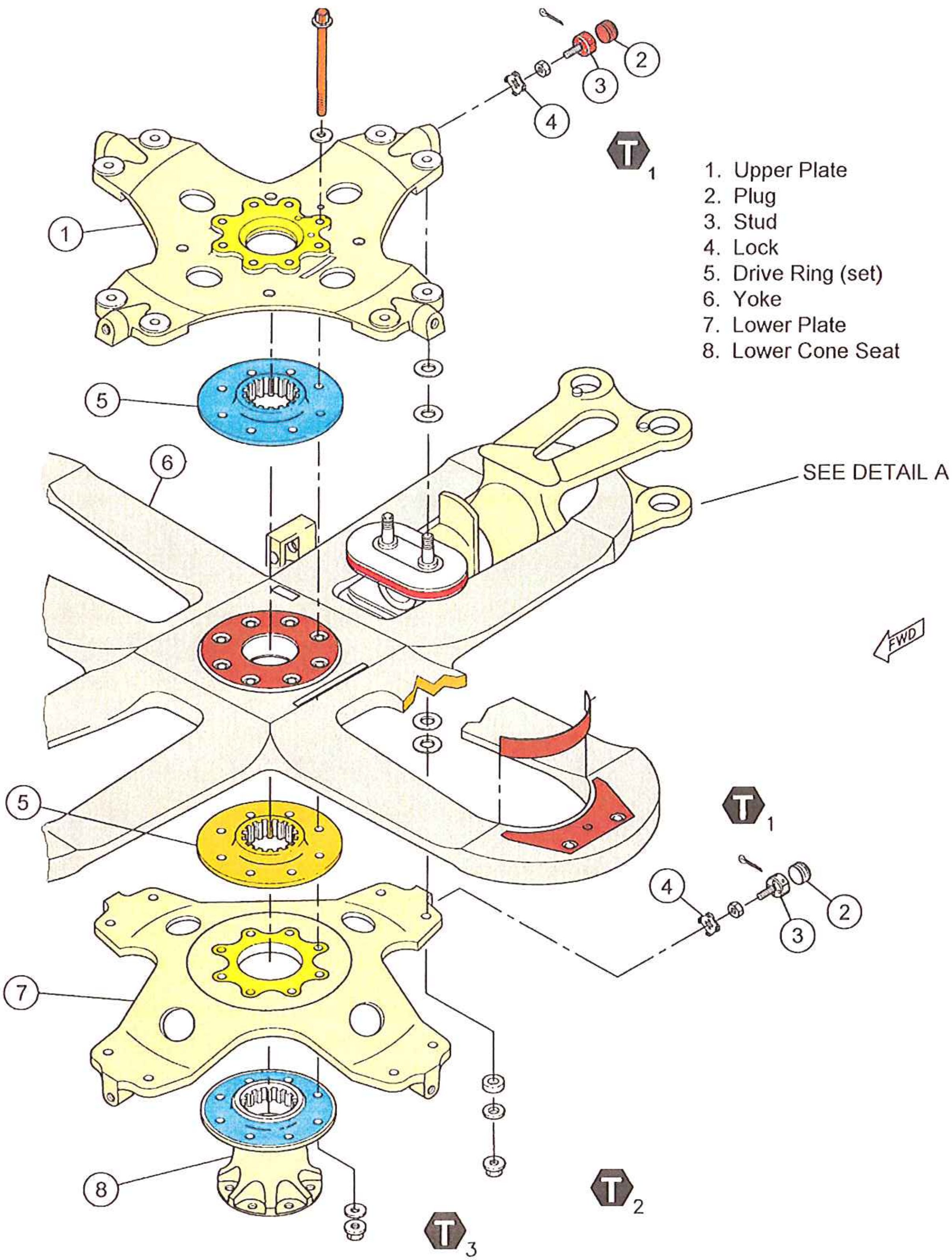

1. Upper Plate
2. Plug
3. Stud
4. Lock
5. Drive Ring (set)
6. Yoke
7. Lower Plate
8. Lower Cone Seat

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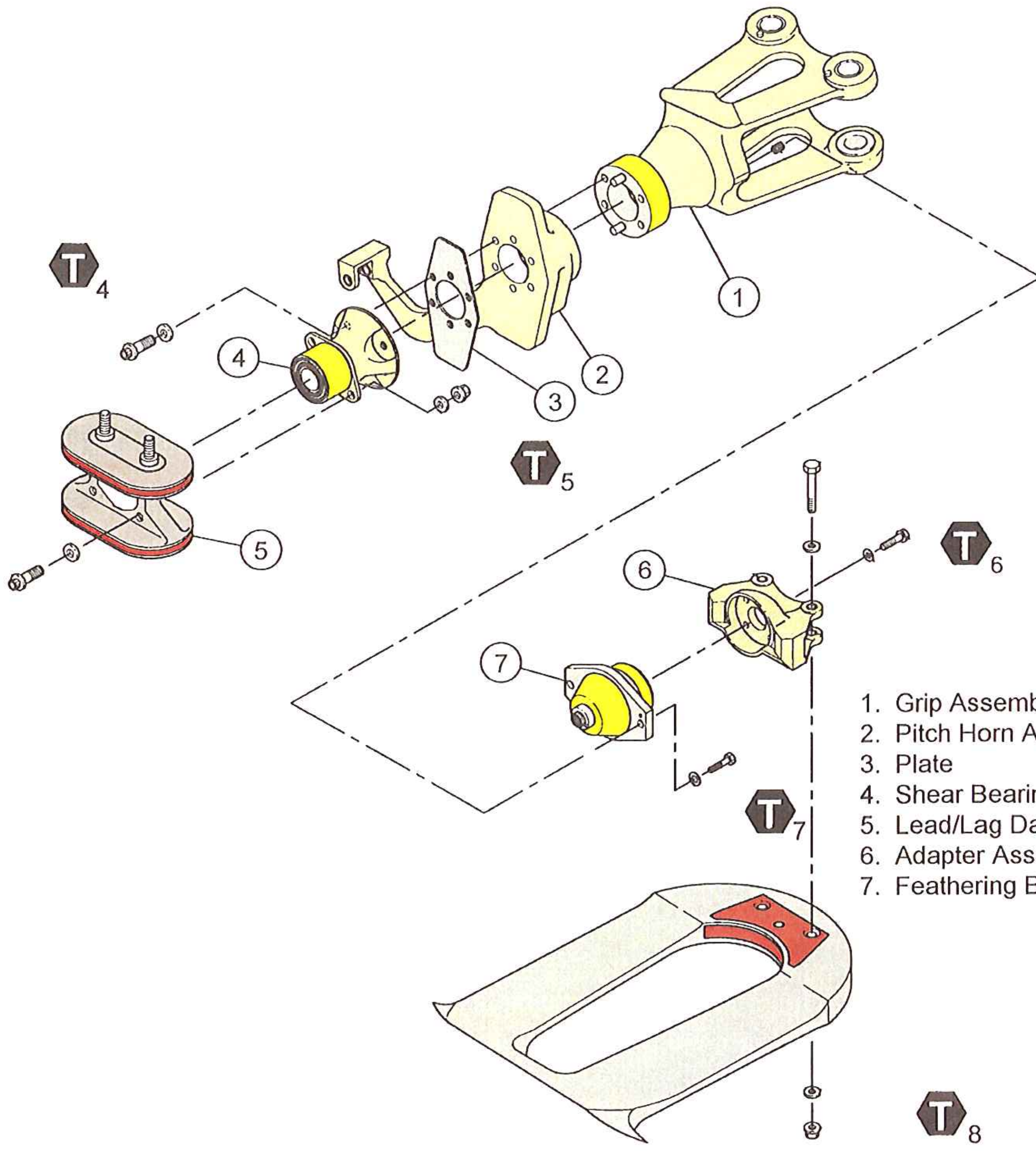

1. Grip Assembly
2. Pitch Horn Assemb
3. Plate
4. Shear Bearing
5. Lead/Lag Damper
6. Adapter Assembly
7. Feathering Bearing

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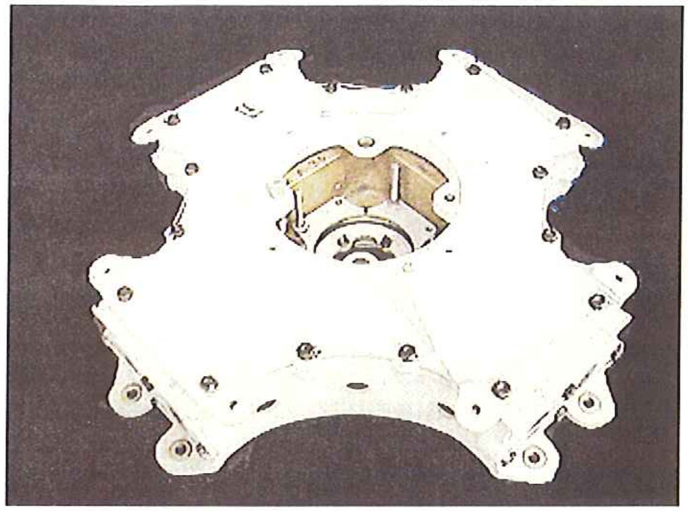
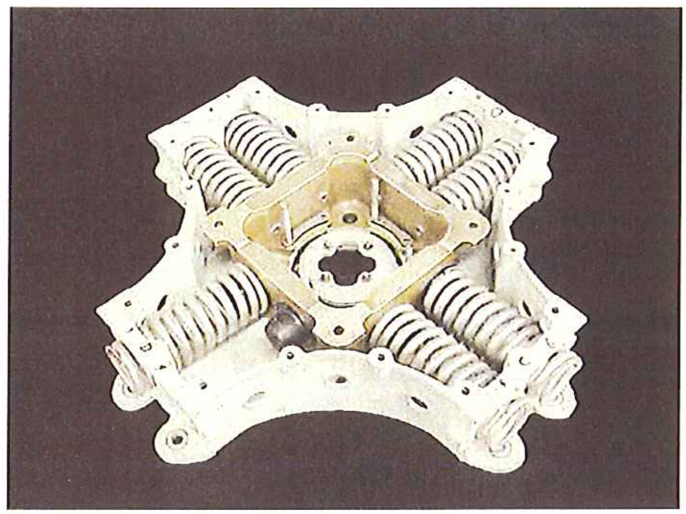
Frahm Damper

A Frahm damper is installed atop the main rotor head to dampen four per rev vibrations from the main rotor and hub

A formed steel block is held in position (floats independent of the upper and lower housing) by 8 heavy gage springs. As the main rotor blade encounters in-flight convective turbulence (frequency), the Frahm damper steel block begins to vibrate at a similar frequency, canceling the vibrations prior to their descent into the main rotor shaft and the aircraft fuselage.





Main Rotor System Cover

A convex fiberglass cover is aerodynamically shaped for drag and noise reduction. The cover is installed on top of the Frahm assembly to protect the Frahm assembly mechanism from any environmental factors.

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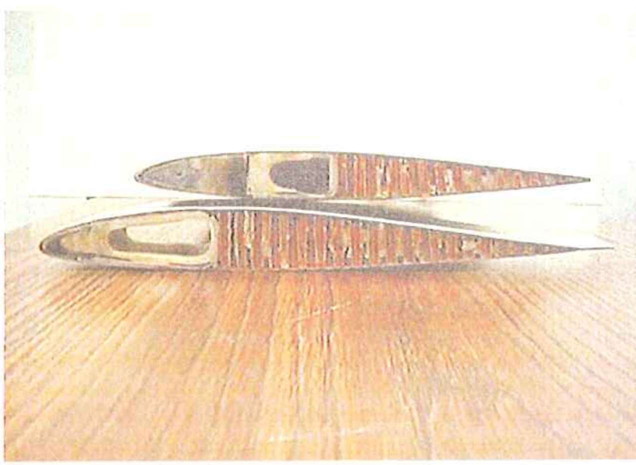
DRIVETRAIN AND ROTORS

Main Rotor Blades

The main rotor blades are asymmetrical for optimum rotor blade efficiency. The blades incorporate a -13° twist from hub to tip to provide a more equal distribution of lift along the length of the blade and to reduce blade stress. The main rotor blades are composite design consisting of three structural members; a fiberglass/epoxy spar, a Nomex honeycomb core, and a fiberglass/epoxy skin. The leading edge is a nickel plated stainless steel strip and is coated with conductive paint to protect the main rotor blade against lightning strikes and static charges. The blades are dynamically balanced and interchangeable.



Product balance weights reduce inherent rotor vibrations during pitch changes.

Tip weights increase rotor inertia for better autorotative characteristics.

Each main rotor blade weighs approximately 58 pounds.

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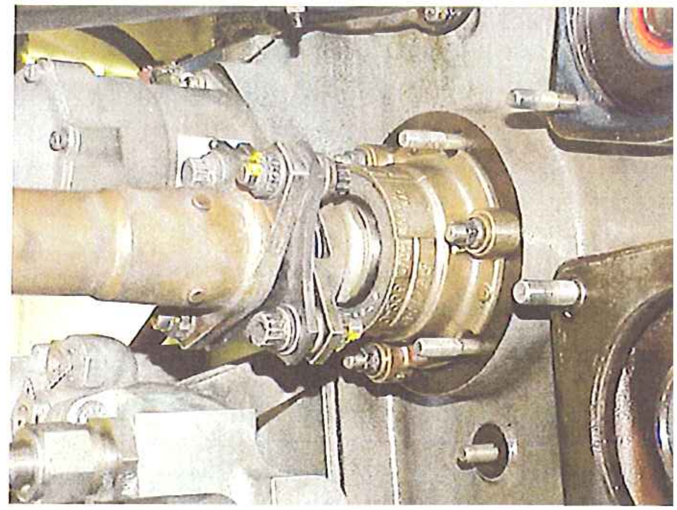
DRIVETRAIN AND ROTORS

Tail Rotor Driveshaft

The segmented tail rotor drive shaft extends along the top of the tailboom. The tail rotor drive shaft is strong, flexible, and segmented to provide power and drive to the tail rotor. The drive shaft consists of several segments. The most forward segment is a steel shaft mounted between the accessory gearbox and the aft firewall. The second shaft segment is a short steel shaft that drives the oil cooler blower.



Aft of the oil cooler blower are five equal length aluminum tail rotor drive shaft segments providing drive into the tail rotor gearbox. Laminated steel disk packs (Thomas Couplings) are used to connect the shaft sections and maintain alignment with the tailboom. The tail rotor driveshaft segments are mounted onto the top of the tailboom utilizing seven hanger bearing assemblies.

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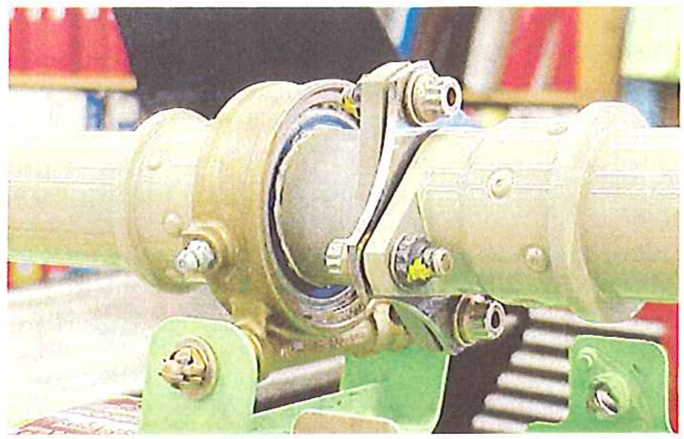
Hanger Bearings

Seven hanger bearing assemblies are utilized to support the tail rotor driveshaft segments. These hanger bearings are grease lubrlcated and assist in maintaining the alignment of the driveshaft along the top of the tailboom.

Thomas Couplings



Eight coupling disc packs (Thomas couplings) are used to connect the tail rotor drive shaft segments together. Each disc pack is made up of 9 to 12 steel plates. Each plate's circumference is cut in two places, parallel to the grain of the steel. When assembled each plate is turned 90° from the previous plate to alternate the grain of the steel plates. This increases the strength and flexibility of the disk pack. Verify only the grip portion (non-threaded section or shoulder) of the bolts are positioned to contact the disc pack.

Tail Rotor Gearbox

The tail rotor gearbox is located on the aft end of the tailboom. The gearbox drives the tail rotor. The direction of drive is changed ninety degrees by means of two spiral bevel gears positioned at a ninety degree angle to the each other. There is a speed reduction of 2.53:1 at the gearbox. The magnesium housing is attached to the tailboom by means of four bolts and two alignment pins. The assembly includes a non-vented filler cap, vent line, oil level sight glass, and a combination magnetic chip detector/self-closing drain valve. The chip detector consists of a self-locking bayonet probe with a permanent magnet at the end. Free ferrous metal particles in the oil are attracted to the magnet. When sufficient metal is attracted to complete the circuit between pole and ground, the T/R CHIP detector segment on the caution panel will illuminate. The valve automatically closes and prevents loss of oil when the chip detector is removed for inspection. The valve also serves as a drain plug when oil needs to be drained from the gearbox.

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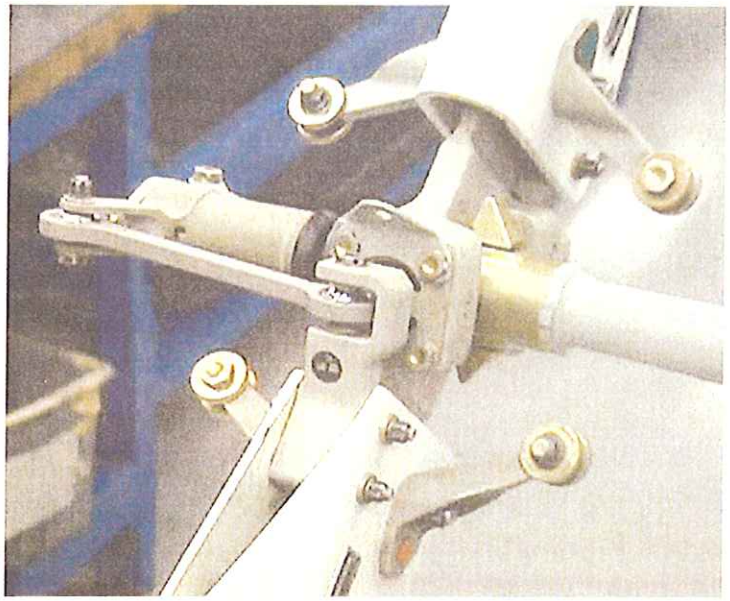
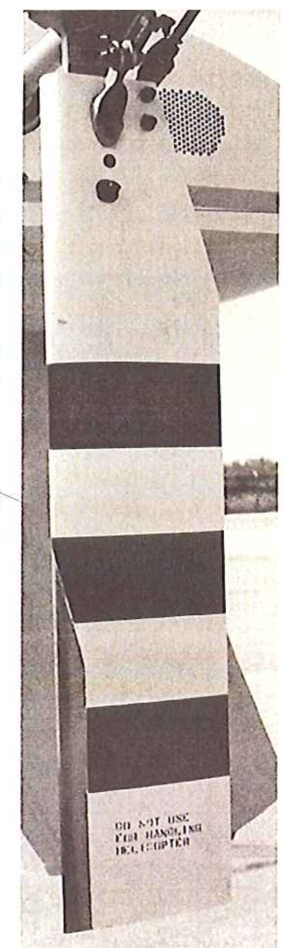
DRIVETRAIN AND ROTORS

Static Stop and Yield Indicator



A static stop and yield indicator is installed on the tail rotor output shaft inboard of the yoke. This static stop and yield indicator provide the ability to visually determine if the tail rotor yoke has been stressed beyond design limits. This will be evident by deformation of the static stop yield indicators due to excessive contact with the yoke. If any deformation of either yield indicator is evident, maintenance action may be required.

Tail Rotor Hub and Blades



The tail rotor is a two bladed teetering rotor with a 5.42 feet diameter. A stainless steel yoke is attached to the output shaft by means of a splined elastomeric bearing trunnion. The trunnion provides a flapping axis for the assembly and compensates for dissymmetry of lift. Two feathering bearings per blade, allow blade feathering. Span-wise balance is accomplished by using washers on the blade bolts at the yoke. Chord-wise balance is accomplished by using weights and washers on the trunnion bearing housing restraining bolts.

The tail rotor yoke fits into the root of the tail rotor blade. Two bolts attach the blade and yoke together. A pitch horn is attached from each blade root to the crosshead to provide angle of attack changes.

The all composite tail rotor blade is made of a fiberglass/epoxy spar and skin assemblies, Nomex honeycomb core, and a nickel plated stainless steel leading edge abrasion strip. The blades are painted with conductive paint for lightning protection and conductivity.

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DRIVETRAIN AND ROTORS

Servicing

TRANSMISSION and TAILROTOR GEARBOX

Oils confirming to the following specifications are approved for use in transmission and tailrotor gearbox.

NOTE

It is recommended that DOD-L-85734 oil be used in transmission and tailrotor gearbox to maximum extent allowed by temperature limitations. Refer to BHT-407-FM-1 for transmission and tailrotor gearbox oil limitations

Sight glasses are provides to determine quantity of oil in transmission and tailrotor gearbox.

NOTE

Transmission oil may partially drain into freewheeling assembly after shutdown. When checking oil levels consider this and slope of helicopter landing surface. If not considered, a false oil quantity indication or overfilling of a gearbox could result.

Transmission capacity	5.0 U.S. quarts (4.7 liters)
Tailrotor Gearbox capacity	0.33 U.S. quarts (0.31 liters)

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Bell Helicopter

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Flight Controls and Hydraulics

Training Academy

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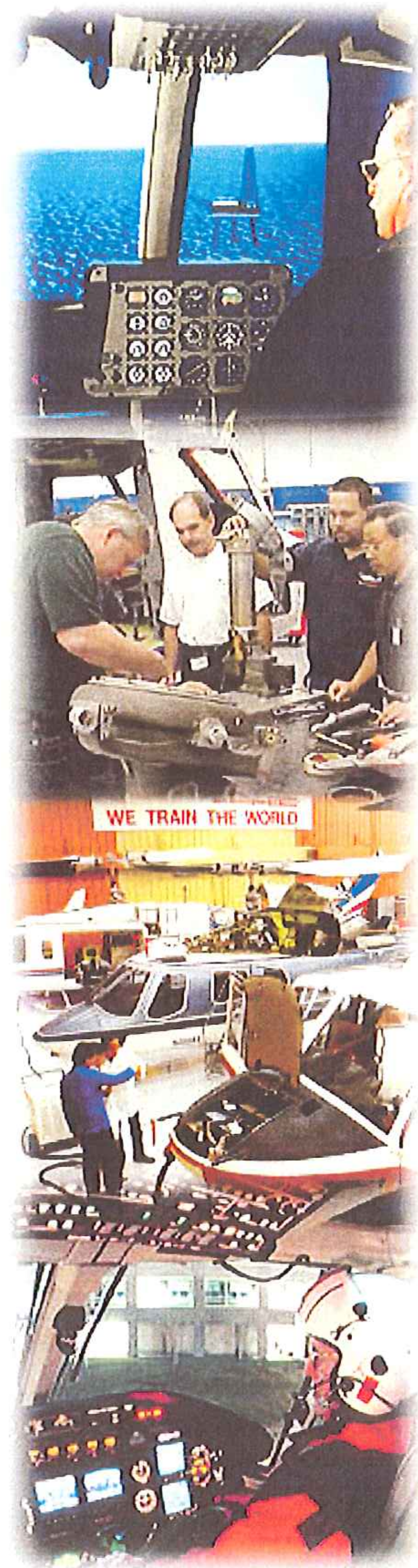

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FLIGHT CONTROLS and HYDRAULICS

Main Rotor Controls

The main rotor and tail rotor flight controls systems consists of cyclic, collective and anti-torque pedals and associated mechanical linkage. These controls are used to regulate the helicopter attitude, altitude and direction of flight. All controls are hydraulically boosted to reduce pilot effort and to counteract control feedback forces.

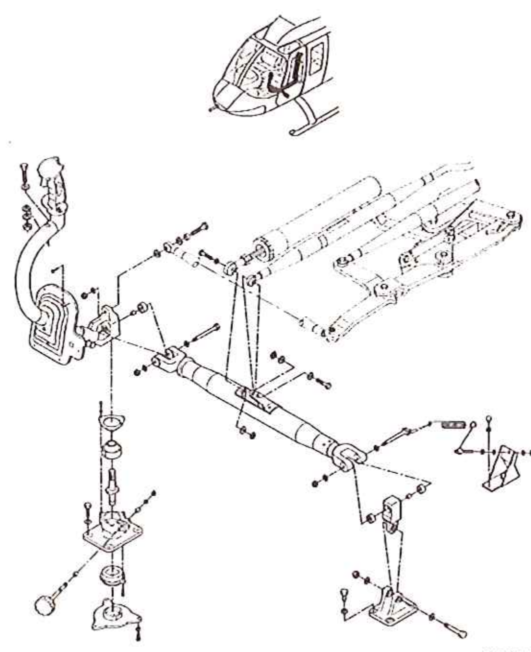
The main rotor cyclic and collective controls regulate pitch / roll attitude and thrust. Control inputs from the cyclic and collective are transmitted by push-pull tubes to their respective hydraulic servo actuator. The actuators are connected to the cyclic and collective levers and will control the elevation and tilt of the swashplate. The swashplate converts linear control inputs from the cockpit to the rotating controls and allows cyclic and collective pitch input to the rotor systems.

Springs are installed in parallel to the cyclic and collective push-pull tubes on the cabin roof to assist the pilot with the increased control feedback forces if hydraulic pressure to the servo actuators is lost.

Cyclic

The cyclic is mounted under the pilots crew seat and protrudes from the forward bulkhead of the crew seat. Fore and aft cyclic input is connected through push-pull tubes to the cyclic hydraulic servo actuators. This input is also fed through a cam assembly that automatically adds lateral cyclic input proportional to the fore and aft cyclic movement. A spring canister is provided in line with the cam input to permit cyclic movement in the event the cam assembly becomes jammed.

The lateral cyclic input is connected through push-pull tubes to the cyclic hydraulic servo actuator. The hydraulic servo actuators operate bellcranks and push-pull tubes that tilt the swashplate non-rotating ring. The swashplate rotating ring will tilt in unison and actuate the pitch links that control the plane of rotation of the main rotor.



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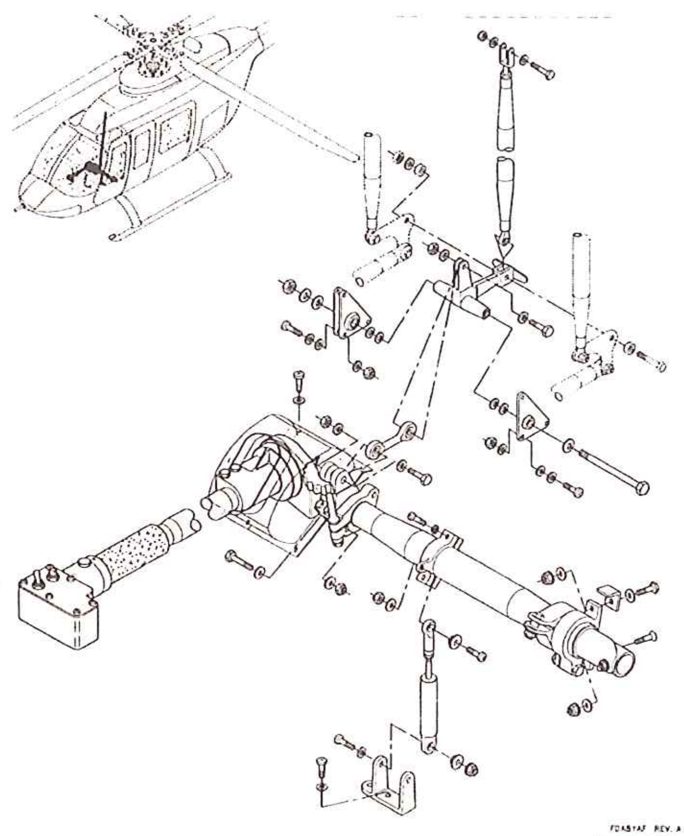
The cyclic grip contains a two-position intercommunication/radio transmit switch and a cargo hook release switch. An adjustable friction control knob located at the base of the cyclic stick allows the pilot to set the desired amount of control stiffness for flight or to lock the cyclic control stick during ground operation or shutdown.

Collective



The collective is mounted between the pilot and the co-pilot crew seats. The collective is connected to the collective hydraulic servo actuator through push-pull tubes. This servo actuator connects to the collective lever mounted on top of the transmission. The collective lever raises and lowers the swashplate ball-sleeve assembly and cyclic levers to induce blade angle change to the main rotor system without affecting the cyclic path. A spring is installed under the copilot's crew seat to balance the force required to raise and lower the collective with the hydraulic boost system operating.

A collective friction knob is located near the base of the collective between the pilot and the co-pilot's seats. A twist grip throttle for the engine is mounted on the collective stick. A mechanical idle release push button is located in front of the twist grip throttle. A switchbox located on the forward end of the collective provides a mount for the engine start switch and landing light switch.

Control Column

The flight controls are routed beneath the pilot's and passenger seats aft to the vertical control column then up to the cabin roof. This control column also serves as a primary cabin support structure. Access panels on the aft side of the column, the bottom of the aircraft, and seat panels are provided for inspection of control components and maintenance accessibility. The cyclic controls are mixed with collective control through the mixing lever located at the base of the control column.

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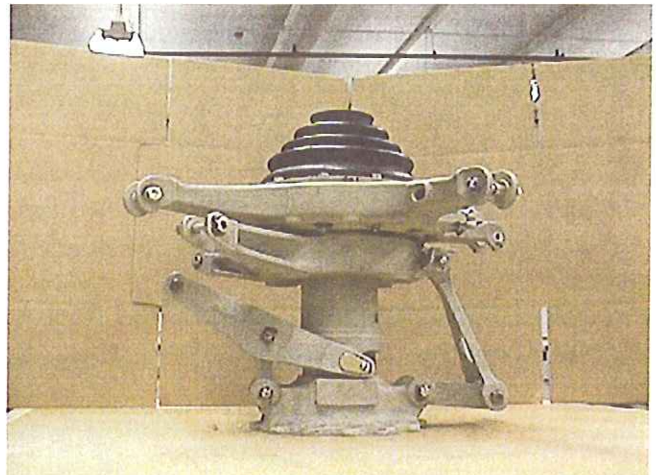
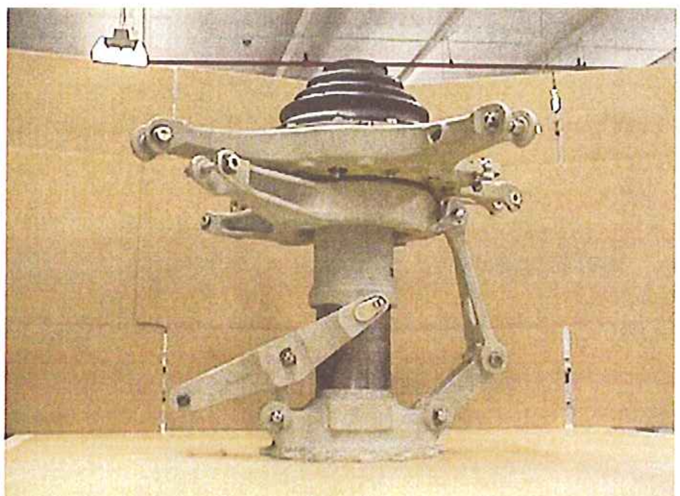
FLIGHT CONTROLS and HYDRAULICS

Swashplate and Collective Lever

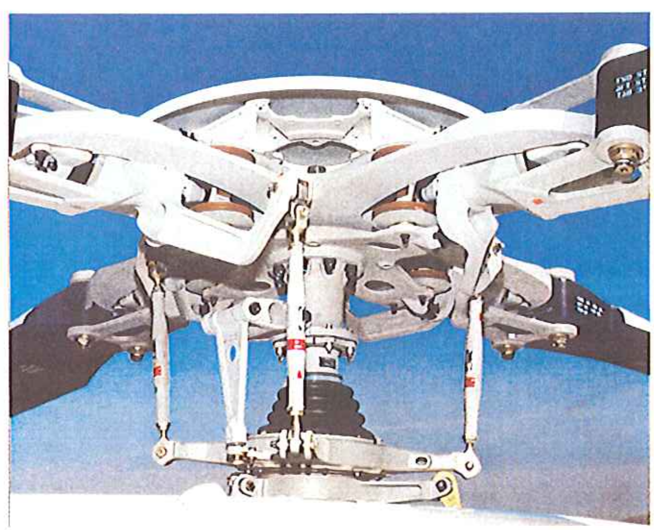
The main rotor controls consist of the swashplate and support assembly, sleeve drive link, and pitch links. The swashplate and support assembly transfers cyclic control motions from the non-rotating to the rotating control system. The swashplate and support encircle the mast directly above the transmission. The swashplate mounts on a universal support (pivot sleeve or uniball) that permits it to move in any direction. Movement of the cyclic results in a corresponding tilt of the swashplate and the main rotor.





The collective lever and link assembly is mounted to the swashplate support assembly and transfers collective inputs to the lower swashplate. Movement of the collective pitch lever actuates the sleeve assembly that in turn raises or lowers the swashplate and transmits collective input to the main rotor. The swashplate drive assembly consists of a collar set, two idler links, and idler levers. The collar set is attached to the mast and the idler links are attached to the outer ring of the swashplate. This connects the upper swashplate to the mast, causing it to rotate with the mast.

The pitch link assembly connects the pitch horn on the blade grips to the swashplate that transmits control input from both the collective and cyclic controls.



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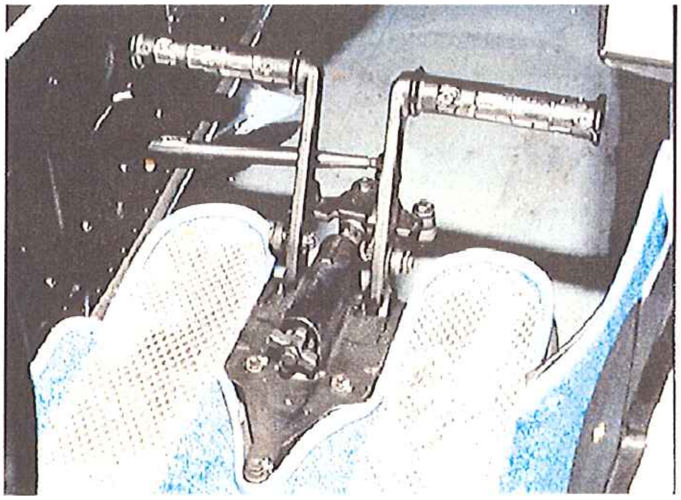
Tail Rotor Controls



The tail rotor controls include the control pedal assembly, pedal adjuster, control tubes, bellcranks, hydraulic actuator, and a pitch control mechanism mounted through the tail rotor gearbox shaft. Moving the pedals causes pitch change in the tail rotor blades to offset the main rotor torque and to control the directional heading of the helicopter.

The tail rotor pedals are connected to a pedal adjuster that provides for manual adjustment of the pedal position according to pilot needs.

Tail rotor pitch control is accomplished by means of bellcrank, rod, and lever assembly mounted on the tail rotor gearbox that actuates a control tube through the hollow output drive shaft to the crosshead and pitch links.

Pedal Stop Solenoid

This solenoid will extend the pedal stop restrictor into the left pedal range of travel. The restrictor will allow full left pedal at less than 50 knots and will limit left pedal travel when airspeed increases to 55 knots and above.

Dual Controls

Installation of dual controls provides a collective, cyclic, and a tail rotor control pedal assembly for the copilot. The copilot's controls are connected to the pilot's by means of the jackshaft, control tubes, and electrical wiring. Quick disconnects are provided for the collective and cyclic. The copilot's controls do not provide electrical cargo release, flight idle stop, throttle bezel, marking, starter switch, or landing light controls.

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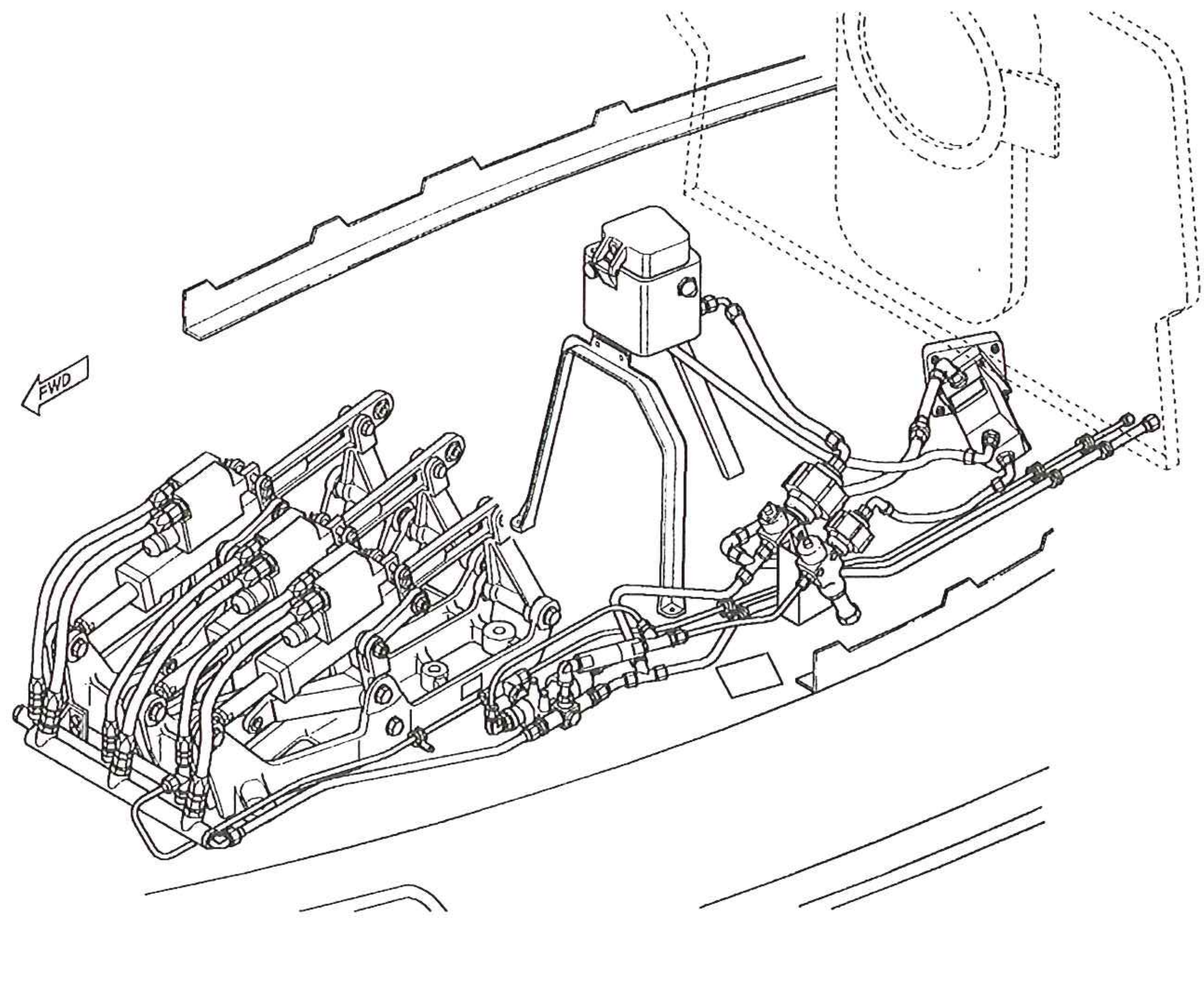
FLIGHT CONTROLS and HYDRAULICS

Hydraulic System General

The hydraulic system provides pressurized fluid to operate the cyclic, collective, and tail rotor flight control servo actuators. Operation of the system is electrically controlled by means of the hydraulic system switch. When the hydraulic system solenoid valve is de-energized (switch on), the pressurized hydraulic fluid flows to the four servo actuators. When it is energized (switch off), the pressurized hydraulic fluid flows to the reservoir and bypasses the four servo actuators. In case of a total electrical failure, the system is fail-safe ON.

The hydraulic reservoir and cover are constructed from magnesium alloy. The reservoir is mounted on a brace and support forward of the main transmission.

The hydraulic system has no specific cooling system. Heat is dissipated from the hoses and lines as air circulates beneath the forward cowling.



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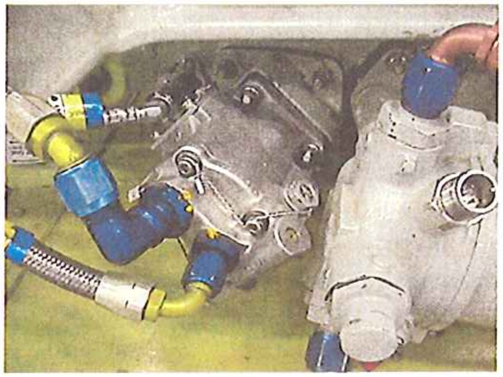
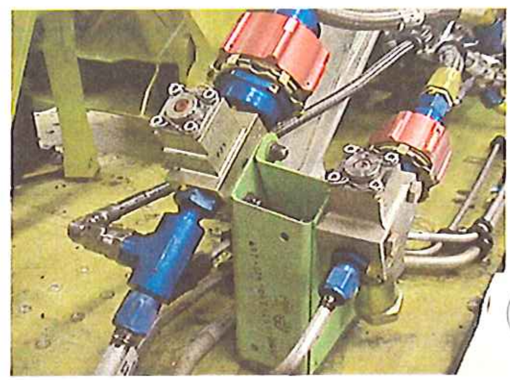
Hydraulic System Components

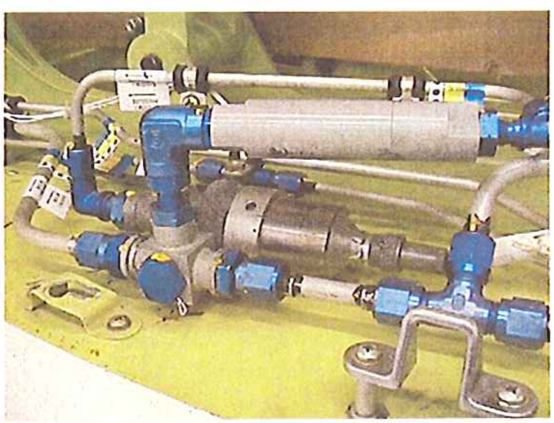


The hydraulic pump is mounted on and driven by the transmission. The hydraulic pump is driven by the transmission accessory drive. The pump is a constant pressure, variable delivery, self-lubricated type designed to operate continuously and provide a rated discharge pressure of 1,000 ± 25 PSI.

Two hydraulic filters are installed on a bracket on the forward left side of the cabin roof near the main transmission. Each filter assembly contains a filter indicator that indicates an impending clogged filter. The indicator consists of a red button mounted on the filter assembly housing. When the differential pressure across the filter is 70±10 PSID the red button will rise. To prevent inaccurate indications of bypass, the indicator will not work when the hydraulic fluid temperature is less than 35° F (2°C). If the hydraulic fluid temperature is more than 35° F (2°C) the indicator gives the correct indication of clogging, even if the ambient (OAT) temperature is below 35° F.



One filter is installed in the pressure line and the other in the return line. The pressure filter assembly does not have bypass capability. The return line filter does incorporate a bypass valve, which serves to prevent over pressurization of the return system. If the filter element in the return line becomes clogged, the bypass valve will open and allow fluid to bypass the filter and return to the reservoir.



The relief valve is installed forward of the transmission and on the left side of the cyclic and collective servo actuator support. During normal operations the relief valve is in the closed position. If system pressure should increase to 1,075 to 1,375 psi, the valve will open to protect the system from damage by returning excess hydraulic fluid to the pump. When system pressure returns to normal the relief valve will close.

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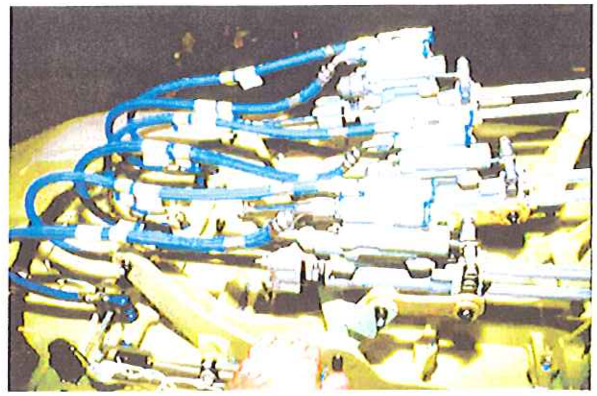
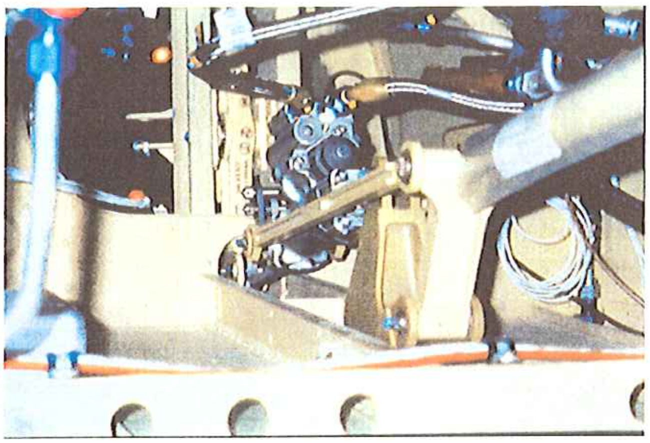
FLIGHT CONTROLS and HYDRAULICS

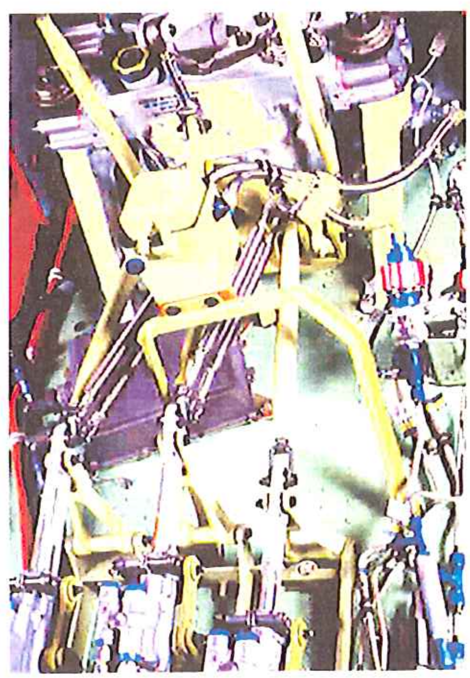
The cyclic and collective servo actuator support is installed on the cabin roof forward of the transmission. It serves as a mount for the three servo actuators and associated bell-cranks. The collective servo actuator is mounted in the center position and the two cyclic servo actuators are mounted on the outboard positions.





The tail rotor servo actuator is mounted in the intermediate section of the fuselage above the baggage compartment.

Springs are installed in parallel to the cyclic and collective push pull tubes on the cabin roof. If hydraulic pressure is lost to the servo actuators, these springs provide assistance to the pilot with the increased control feedback forces.



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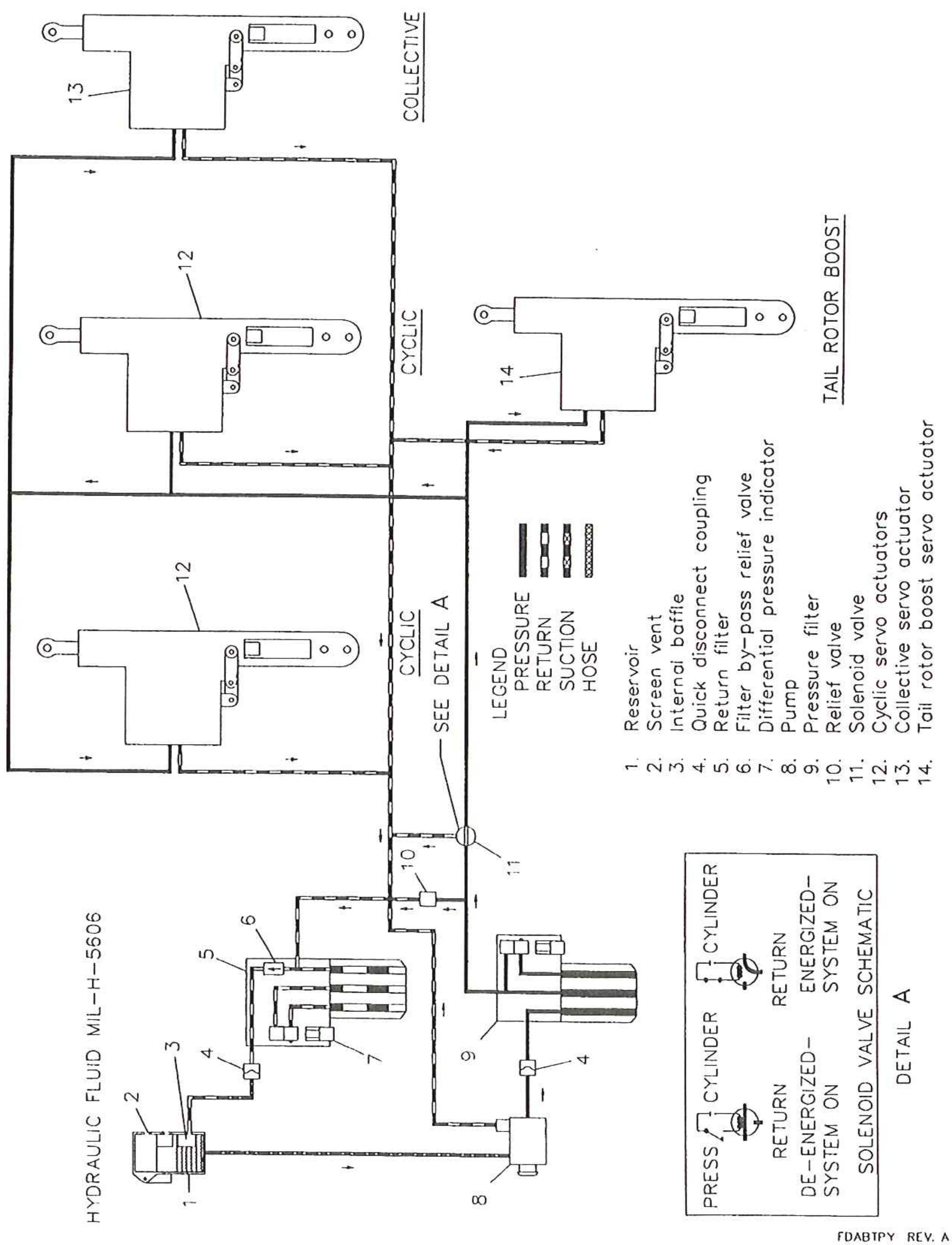

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FLIGHT CONTROLS and HYDRAULICS

Hydraulic Systems Check

(Modified Excerpt from M407 Flight Manual)

PRELIMINARY HYDRAULIC SYSTEMS CHECK

NOTE

Uncommanded control movement or motoring with hydraulic system off may indicate hydraulic system malfunction.

HYDRAULIC SYSTEM switch — OFF, caution light illuminated, switch — ON.

HYDRAULIC SYSTEMS CHECK

NOTE

HYDRAULIC SYSTEMS CHECK is to determine proper operation of hydraulic actuators for each flight control system. If abnormal forces, unequal forces, control binding, or motoring are encountered, it may be an indication of a malfunctioning flight control actuator.

Collective — Full down.

ROTOR — 100% RPM.

HYDRAULIC SYSTEM switch — OFF, caution light illuminated.

Cyclic — Centered.

Cyclic control -Check normal operation by moving cyclic forward and aft , then left and right (approximately 1 inch) center cyclic.

Collective — Check for normal operation by increasing collective slightly (1 to 2 inches). Repeat 2 to 3 times as required. Return to full down position.

Pedals - Check normal operation by displacing pedals slightly (1 inch).

HYDRAULIC SYSTEM switch — ON, caution light extinguished.

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Servicing

Hydraulic fluids conforming to MIL-H-5606 (NATO H-515) and are approved for use in hydraulic flight control system and rotorbrake.

Hydraulics Reservoir Capacity: 1.0 U.S. pint (0.5 liter).

Hydraulic reservoir is located on top of fuselage, forward of transmission and under forward faring. A sight glass is provided to determine quantity of hydraulic fluid in reservoir.

Service hydraulic system as follows:

1. Open and support top of forward fairing.

2. Remove cap and fill reservoir until sight glass is full of hydraulic fluid.

3. Secure cap and fairing.

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Weight and Balance/Performance

Training Academy

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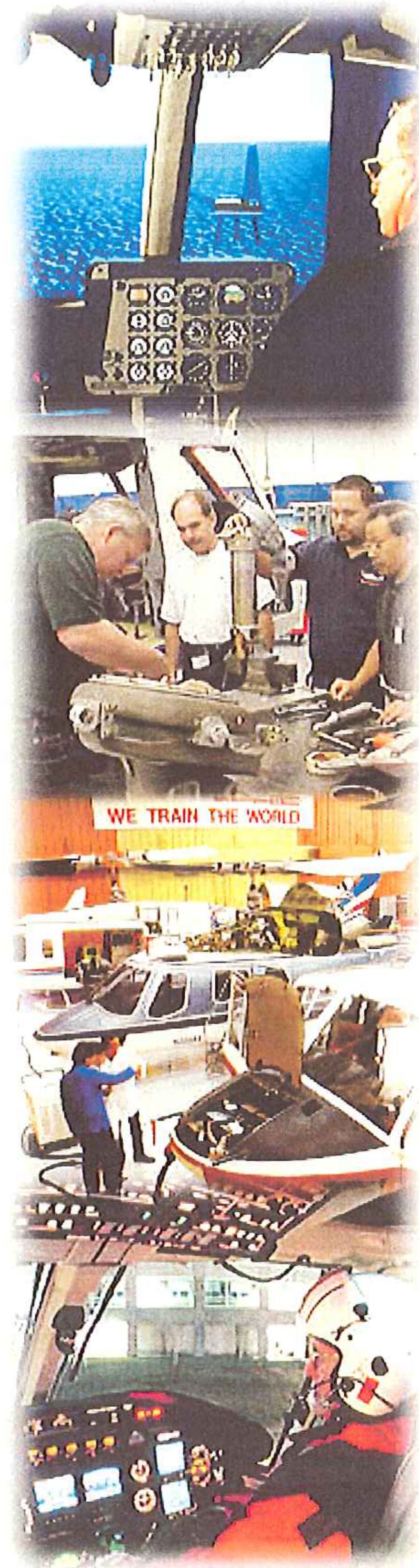

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WEIGHT AND BALANCE/PERFORMANCE

General Weight and Balance

Section 5 of the approved rotorcraft flight manual (RFM) for the Bell 407 presents the data necessary for the pilot to compute gross weight and center of gravity for various load configurations. It is the pilot's responsibility to ensure that the maximum gross weight and center of gravity limitations are observed throughout each planned flight. Operations outside of the limitations are prohibited and may result in a reduction of aircraft performance, handling qualities, stability, and structural integrity.

Changes in aircraft configuration (example: doors on or off), loading, seating of passengers, positioning of cargo, and fuel consumption are all factors that must be considered in weight and balance calculations.

Terms

The following terms are used in the calculation of weight and balance information:

Weight: The actual aircraft weight, weight of crewmembers and passengers, fuel, etc. is required to accurately compute weight and balance. The flight manual provides data based on either U.S. or metric measurements, expressed in pounds or kilograms.

Arm: When calculating weight and balance information, this term refers to the distance from a known point to where new weight is added. For a given quantity of weight added, the greater the arm, the greater the affect upon the balance point. In the 407 flight manual, this data is referred to as fuselage stations or buttock lines. It is provided in both U.S. and metric formats (inches or millimeters) and is provided for both longitudinal and lateral axes of balance.

Moment: A mathematical expression of the affect that weight (at a certain position) will have on the balance point. This number is obtained by multiplying the weight times the arm. The result is referred to as moment, and can be expressed as inch-pounds (U.S.) or kilograms-millimeters (metric).

Center of Gravity: Used to identify the position at which the aircraft is in balance. Maintenance personnel compute the Empty Weight Center of Gravity. Pilots use the empty weight CG to determine the Gross Weight Center of Gravity. Total Moment divided by total weight equals the center or gravity. The result is expressed in either inches (U.S.) or millimeters (metric).

Datum: An arbitrary point from which to measure fuselage station (arm) for longitudinal CG calculations. Fuselage Stations aft of the datum line are expressed as positive numbers, while points forward would be expressed as negative numbers.

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WEIGHT AND BALANCE/PERFORMANCE

Center line: A reference point for measuring distances from the lateral axis of the aircraft. Buttock lines are measured from the center line, either to the left or to the right. Positions to the right of the center line are considered positive numbers, while positions to the left are considered negative numbers.

Empty Weight and Center of Gravity: computed by maintenance personnel. The empty weight configuration consists of the basic helicopter with required equipment, operational and special equipment, fixed ballast, hydraulic fluid, transmission and gearbox oil, unusable fuel, and undrainable engine oil. Weight empty and center of gravity is recorded on the Actual Weight Record, a copy of which should be carried in the helicopter for use in weight and balance calculations. Each time equipment is installed or removed, the actual weight record should be updated.

Gross Weight Center of Gravity: computed by the pilot. Gross weight includes the total weight of the helicopter, with contents. Contents include crew, passengers, engine oil, fuel, baggage, and cargo.

Zero fuel weight (ZFW): A calculated value used as a starting point to calculate changes in CG caused by fuel consumption during flight. ZFW includes helicopter empty weight, crew, passengers, engine oil, baggage and cargo.

Effects of Fuel Consumption on Center of Gravity

Under normal circumstances, a center of gravity calculation completed prior to takeoff will be affected only by fuel consumption during the flight. Other loading considerations (crew, passengers, and cargo) will not change.

The automatic fuel transfer sequence will affect the aircraft center of gravity as fuel is consumed. Generally speaking, if the aircraft begins a flight with full fuel and lands with minimum fuel, the center of gravity will move forward, aft, then finally forward. It is the pilot's responsibility to ensure that the helicopter remains within CG as fuel is consumed. To accomplish this, fuel shifts must be calculated. Data is provided in fuel loading tables to compute moment values for various quantities of fuel.

Baggage Compartment Loading

The baggage compartment is accessible from the left side of the fuselage and contains approximately 16 cubic feet (0.45 cubic meters) of space. It has a load limit of 250 pounds (113.4 kg), not to exceed 86 pounds per square foot (4.2 kilograms/100 square centimeters). These are structural limitations only, and do not infer that the CG will remain within approved limits. When weight is loaded into the baggage compartment, the pilot must compute gross weight and CG to assure loading within approved limits. The load shall be secured to tie-down fittings if shifting of the load in flight could result in structural damage to the baggage compartment or in gross weight center of gravity limits being exceeded. Tables and examples assume items in the baggage compartment have a longitudinal CG at the midpoint of the door opening

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WEIGHT AND BALANCE/PERFORMANCE

Weight and Balance Calculation

As previously stated, it is the pilot's responsibility to ensure that the maximum gross weight and center of gravity limitations are observed throughout each planned flight. In the case of the 407, improper cabin loading and fuel consumption may create a situation where the center of gravity could travel outside CG limits during the flight. For this reason, it is important to load the helicopter as evenly as possible, both longitudinally and laterally.

Weight and Balance Exercise

The following pages include several blank weight and balance worksheets that you may use to complete this exercise and for any calculations necessary during the performance planning exercise. (The same weight and balance computation will also be used during the performance planning exercise).

Today, for purposes of this exercise, you are a pilot for the ABC Corporation, a mining engineering firm with business throughout the region. In today's flight, the mission is to transport the Company President, Vice-President of Operations, Vice-President of Human Resources, and the Chief of Engineering from company headquarters at Point Xray to a luncheon engagement at LZ Yankee, then to the international airport at City Zulu to connect with a commercial flight.

Your aircraft is configured with standard seating/standard fuel (no aux tank) and you have obtained the following information from the actual weight record in the Rotorcraft Flight Manual.

Aircraft empty weight: 2967
Longitudinal Moment: 390503
Lateral Moment: -52

Your weight 170 lbs

Passenger Weights

Pres	300 lbs (prefers to sit in the right aft seat)
VP OPS	210 lbs
VP HR	180 lbs
CH ENG	235 lbs

Baggage – 100 lbs of baggage.

Calculate the zero fuel weight, takeoff center of gravity, most forward CG condition center of gravity, and most aft CG condition center of gravity for this anticipated flight.

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WEIGHT AND BALANCE/PERFORMANCE

Weight and Balance Worksheet

		Longitudinal		Lateral	
	Weight (lb)	FS (in)	Moment (lb-in)	BL (in)	Moment (lb-in)
Weight Empty					
Oil	13	205.0	2665	0	
+Pilot		65.0		+14	
+Forward passenger		65.0		-11.1	
+ Mid passenger(s) Left		91.0		-13.0	
Right		91.0		+15.5	
+Aft passenger(s) Left		129.0		-16.8	
Center		129.0		0.0	
Right		129.0		+16.8	
+Baggage		174.0		0.0	
Zero Fuel Weight					
+Fuel				0.0	0
Gross takeoff weight (GTOW)					
Zero Fuel Weight					
+Critical fuel (Most FWD)	508.6		58998	0.0	0
Most forward CG condition					
Zero Fuel Weight					
+Critical Fuel (Most AFT)	193.1		26455	0.0	0
Most AFT CG condition					

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WEIGHT AND BALANCE/PERFORMANCE

Weight and Balance Worksheet

		Longitudinal		Lateral	
	Weight (lb)	FS (in)	Moment (lb-in)	BL (in)	Moment (lb-in)
Weight Empty					
Oil	13	205.0	2665	0	
+Pilot		65.0		+14	
+Forward passenger		65.0		-11.1	
+ Mid passenger(s) Left		91.0		-13.0	
Right		91.0		+15.5	
+Aft passenger(s) Left		129.0		-16.8	
Center		129.0		0.0	
Right		129.0		+16.8	
+Baggage		174.0		0.0	
Zero Fuel Weight					
+Fuel				0.0	0
Gross takeoff weight (GTOW)					
Zero Fuel Weight					
+Critical fuel (Most FWD)	508.6		58998	0.0	0
Most forward CG condition					
Zero Fuel Weight					
+Critical Fuel (Most AFT)	193.1		26455	0.0	0
Most AFT CG condition					

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WEIGHT AND BALANCE/PERFORMANCE

Weight and Balance Worksheet

		Longitudinal		Lateral	
	Weight (lb)	FS (in)	Moment (lb-in)	BL (in)	Moment (lb-in)
Weight Empty					
Oil		205.0	2665	0	
+Pilot		65.0		+14	
+Forward passenger		65.0		-11.1	
+ Mid passenger(s) Left		91.0		-13.0	
Right		91.0		+15.5	
+Aft passenger(s) Left		129.0		-16.8	
Center		129.0		0.0	
Right		129.0		+16.8	
+Baggage		174.0		0.0	
Zero Fuel Weight					
+Fuel				0.0	0
Gross takeoff weight (GTOW)					
Zero Fuel Weight					
+Critical fuel (Most FWD)	508.6		58998	0.0	0
Most forward CG condition					
Zero Fuel Weight					
+Critical Fuel (Most AFT)	193.1		26455	0.0	0
Most AFT CG condition					

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WEIGHT AND BALANCE/PERFORMANCE

Performance Planning - General

The performance section of the Rotorcraft Flight Manual contains the Bell 407 performance information and related charts. All performance charts are based on an engine meeting minimum Rolls-Royce specifications. The data shown is derived from actual flight tests and are intended to provide information to be used in conducting flight operations. This performance data is applicable to the 250C47B engine.

NOTE: The 407 basic configuration does not include a particle separator. If the aircraft has a particle separator installed, the correct performance charts are located in FMS-3. This supplement is not included in the training manual.

FMS-28 is the supplement that allows increased gross weight operation. When operating at takeoff gross weights over 5000 lbs, performance data from FMS-28 should be used. Data from this supplement is included in the training manual in Appendix F, beginning on page F-1.

Power Assurance Check

A power assurance check chart is provided for the Rolls-Royce 250C47B engine. This chart indicates the maximum allowable MGT for an engine that meets minimum specifications. The engine must develop the required torque without exceeding chart MGT in order to meet performance data contained in this section. In this training manual, this chart is located on page D-5.

To perform the power assurance check, turn off all sources of bleed air (anti-ice, particle separator purge, and heater). Establish level flight at an airspeed of 85 to 105 KIAS or Vne, whichever is lower. The check may also be conducted in a hover, depending on ambient conditions and gross weight.

With airspeed established, at sufficient altitude, record the following:

Hp
OAT
MGT
Torque

If the actual MGT value recorded is less than or equal to the chart MGT value, the engine meets minimum power specifications.

Density Altitude

A density altitude chart is provided to aid in the calculation of performance and limitations. Density altitude (Hd) is defined as pressure altitude Hp) corrected for non-standard temperature. Pressure, temperature, and humidity determine air density. Hd is an expression of the density of air in terms of height above sea level; hence the less dense

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the air, the higher the Hd. For standard conditions of temperature and pressure, Hd is the same as Hp. As temperature increases above standard for any altitude, Hd will also increase to values higher than Hp. The chart can also be used to compute a multiplication factor to determine true airspeed. In this training manual, this chart is located on page D-6.

To use the chart, enter from the known temperature at the bottom of the chart. Proceed vertically until intercepting the Hp line. Proceed left to determine Hd, and read right to determine the true airspeed conversion factor.

Height Velocity Envelope

The height velocity envelope charts define the conditions from which a safe landing can be made on a smooth, level, firm surface following an engine failure. For purposes of discussion, it is also important to note that the chart is based on zero wind conditions.

Two other important factors that affect autorotational performance are gross weight and density altitude. In the model 407, an Altitude vs. Gross Weight chart is used to select one of four possible H-V diagrams. For a given outside ambient air temperature, pressure altitude, and gross weight the appropriate limiting envelope can be determined (Region A, B, C, or D). In this training manual these charts can be found beginning on page D-7.

To use the envelopes, begin on the Altitude vs. Gross Weight chart. Enter the chart at the appropriate ambient temperature, move up vertically until intercepting the pressure altitude, then move right until intercepting the appropriate gross weight line. The intercept point determines which envelope the aircraft is operating in. No interpolation is allowed.

Next, select the appropriate H-V envelope diagram (Region A, B, C, or D) and enter the chart either at a known skid height above the ground or a known airspeed. Entering the chart at a known skid height , then proceeding horizontally to intercept the avoid area envelope will calculate minimum airspeed. Entering at a known airspeed, then proceeding vertically to intercept the avoid area envelope will calculate minimum skid height.

Hover Ceiling

Hover Ceiling In Ground Effect (IGE) and Out of Ground Effect (OGE) charts present hover performance expressed as allowable gross weight for varying conditions of Hp and OAT. These hovering weights are obtainable in zero wind conditions. The hover performance charts are based on 100% Rotor RPM.

Each hover ceiling chart is divided into two areas. Area A is non-shaded and presents hover performance for conditions where adequate control margins exists for all relative wind conditions up to 35 knots (lateral CG not exceeding ±2.5 inches) or up to 17 knots (lateral CG not exceeding ±4.0 inches). Area B is shaded and presents hover performance for relative winds within ±45 degrees of the nose of the helicopter up to 35 knots (lateral CG not exceeding ±2.5 inches) or up to 17 knots (lateral CG not exceeding ±4.0 inches).

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WEIGHT AND BALANCE/PERFORMANCE

Satisfactory stability and aircraft control have been demonstrated in each area of the hover ceiling charts with winds as depicted on the hover ceiling wind accountability chart. In the training manual, these charts are presented beginning on page D-12.

To use the chart, enter at the appropriate OAT, and proceed vertically to the applicable Hp line. Move horizontally to the right until intercepting the appropriate OAT line. Move vertically down and read maximum gross weight for hover.

Rate of Climb

Rate of climb charts are presented for various combinations of power settings and can be adjusted for doors off and anti-ice ON or OFF. Rate of climb data sown are "tapeline" rates, which means actual rates of climb. Rate of climb as measured with an altimeter will equal this rate of climb only on a standard day with a standard temperature lapse rate.

The rate of climb is reduced 135 feet per minute above 300 feet pressure altitude with anti-ice ON. Reduce rate of climb 100 feet per minute when operating with any combination of door(s) off.

Rate of climb charts are based on a recommended airspeed for best rate of climb 60 KIAS. In the training manual, these charts are presented beginning on page D-30.

To use the charts, select the appropriate chart based on gross weight and enter at the Hp value. Proceed horizontally to the right until intercepting the OAT. Proceed vertically down and read rate of climb.

Descent (Autorotation)

The descent chart for autorotation presents autorotation glide distance as a function of altitude information. The chart is valid for two airspeeds, either minimum rate of descent airspeed (55 knots) or maximum glide airspeed (80 knots). In the training manual, this chart is located on page D-50.

To use the chart, enter at the appropriate Hp. Proceed horizontally to the right until intercepting either the 55 knots or 80 knots airspeed line. Proceed vertically down and read distance over the ground.

Airspeed Calibration

The airspeed calibration table presents calibrated airspeed for various indicated airspeeds. Calibrated airspeed is defined as indicated airspeed corrected for installation errors. Installation error changes based on flight condition, therefore calibrated airspeed values are presented for climb conditions as well as level flight. In the training manual, this chart is located on page D-51.

To use the table, enter at the appropriate KIAS value, then read KCAS from either column.

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WEIGHT AND BALANCE/PERFORMANCE

Performance Planning Exercise

This exercise is designed to practice selecting and using the proper performance charts to predict aircraft performance and mission capability. Weight and balance data presented in the weight and balance exercise is repeated here for use in this exercise.

Today, for purposes of this exercise, you are a pilot for the ABC Corporation, a mining engineering firm with business throughout the region. In today's flight, the mission is to transport the company president, vice-president of operations, vice-president of human resources, and the chief of engineering from company headquarters Point Xray to a luncheon engagement at LZ Yankee, then to the international airport at City Zulu to connect with a commercial flight.

Aircraft Configuration: Standard seating, standard fuel (no aux tank), basic inlet (no particle separator), with all doors ON.

Weight and Balance Data:
Aircraft empty weight: 2967
Longitudinal Moment: 390503
Lateral Moment: -52

Your weight 170 lbs

Passenger Weights

Pres	300 lbs (prefers to sit in the right aft seat)
VP OPS	210 lbs
VP HR	180 lbs
CH ENG	235 lbs

Baggage – 100 lbs of baggage.

Flight Plan: Depart Point XRAY at 1000 CST and proceed to LZ Yankee with arrival at 1210 CST. Depart LZ Yankee at 1400 CST and proceed to City Zulu airport with arrival 1430 CST.

Location Information:

Point XRAY: Elevation 2000 MSL. Winds for the day expected 360 at 20K Temp at departure +26, max expected +35

LZ Yankee (data from airport close by) Elevation 6000 MSL. Winds for the day expected 270 at 20K Temp at arrival +26, max expected +32

City Zulu: Elevation 3000 MSL. Winds for the day expected 240 at 10K Temp at arrival +30, max expected +32

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WEIGHT AND BALANCE/PERFORMANCE

Note: Predicted fuel flow chart is available on page 9-12, if necessary.

Answer the following questions.

1. What is the density altitude at Point XRAY at departure? ____________

2. You elect to maintain 110 KIAS at your anticipated cruise altitude of 6500 feet. What is your anticipated True Airspeed (TAS)? _______________

3. During climbout to your expected cruise altitude of 6500 feet, what rate of climb can you expect passing through 4000 feet with 60 KIAS? ____________

4. Arriving at your cruise altitude (6500 feet) you elect to perform a power check to verify landing performance at LZ Yankee. You establish an airspeed of 105 KIAS, a torque value of 70%, and note an actual MGT reading of 729 degrees centigrade. Is the engine producing acceptable power?

5. How much fuel do you estimate will be consumed during the flight to LZ Yankee?

6. You have never landed before at LZ Yankee, so you are planning for Hover OGE capability. Considering your gross weight at arrival and estimated landing conditions at LZ Yankee, what is the maximum gross weight for hover out of ground effect?

7. Arriving at LZ Yankee, you determine that you will need to refuel at a nearby airport to complete the mission. You decide to drop off the passengers for lunch, then fly to the airport nearby for fuel. Considering gross weight and center of gravity restrictions, is this flight possible? If not, why not? ___________________________. How much fuel may be added to the aircraft considering takeoff conditions from LZ Yankee? ______________

8. While waiting for your passengers to complete their lunch, the winds at LZ Yankee increase to 25 knots gusting to 35. Due to obstacles around the LZ the takeoff must be made to the North. What is the maximum gross weight for takeoff if IGE capability is needed?

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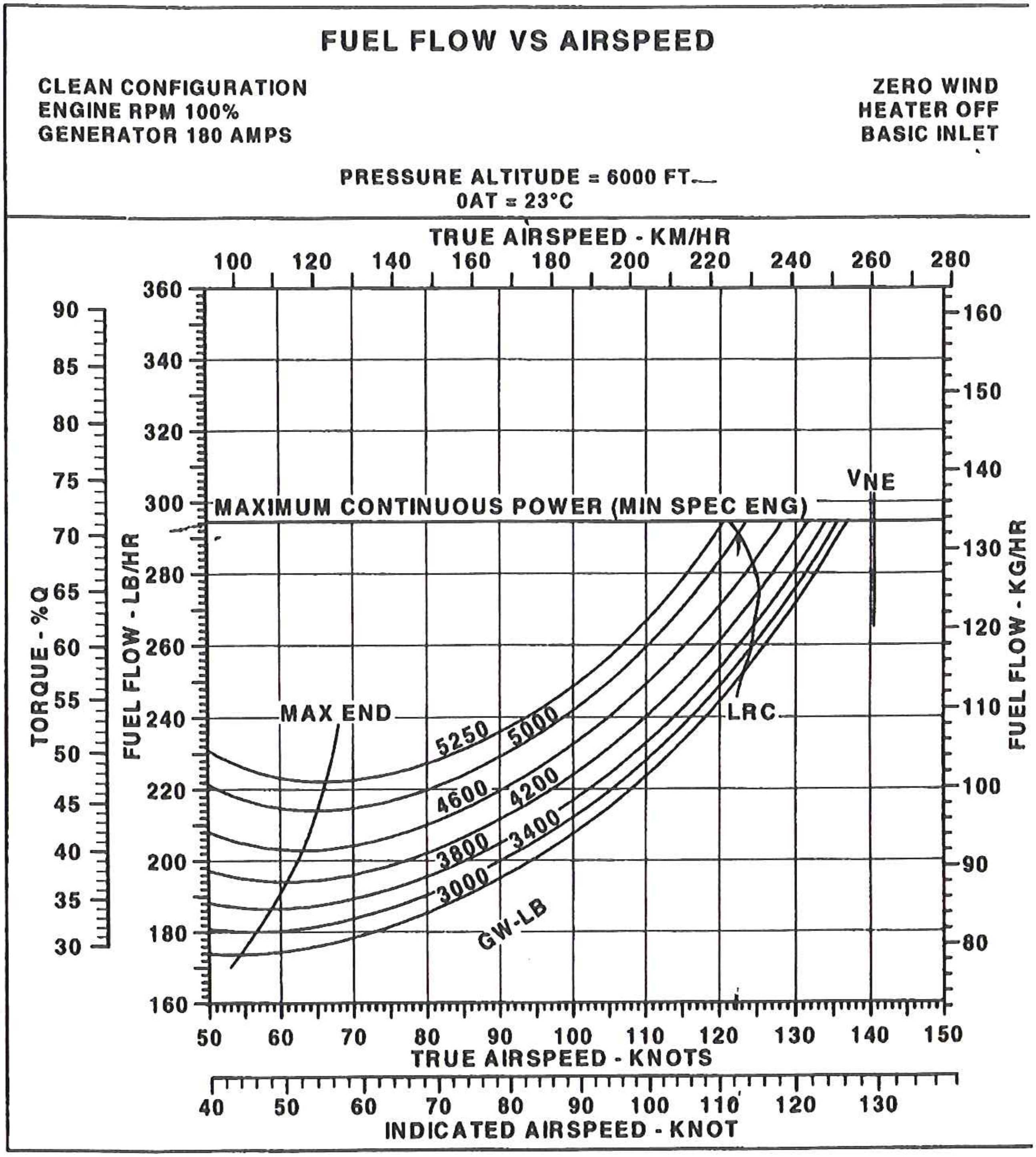

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Section 1

LIMITATIONS

1-1. INTRODUCTION

Compliance with limitations section is required by appropriate operating rules. Anytime an operating limitation is exceeded, an appropriate entry shall be made in helicopter logbook. Entry shall state which limit was exceeded, duration of time, extreme value attained, and any additional information essential in determining maintenance action required.

Intentional use of transient limits is prohibited.

Torque events shall be recorded. A torque event is defined as a takeoff or lift, internal or external load (BHT-407-MD-1).

1-2. BASIS OF CERTIFICATION

This helicopter is certified under FARs Part 27 and 36, Appendix J. Additionally, it is approved under Canadian Airworthiness Manual Chapters 516 (ICAO Chapter 11) and 527, Sections 1093 (b) (1) (ii) and (iii), 1301-1, 1557 (c) (3), 1581 (e) and 1583 (h).

1-3. TYPES OF OPERATION

1-3-A. PASSENGERS

Basic configured helicopter is approved for seven place seating and is certified for land operation under day or night VFR nonicing conditions.

1-3-B. CARGO

The maximum allowable cabin deck loading for cargo is 75 pounds per square foot (3.7 kilograms per 100 square centimeters). The maximum allowable baggage compartment deck loading is 86 pounds per square foot (4.2 kilograms per 100 square centimeters) with a maximum allowable weight of 250 pounds (113.4 kilograms). Refer to BHT-407-MD-1 for cargo restraint and tiedown locations.

Cargo must be properly secured by tiedown devices to prevent the load from shifting under anticipated flight and ground operations. If the mission requires both passengers and cargo to be transported together, the cargo must be loaded and secured so that it does not obstruct passenger access to exits.

1-4. FLIGHT CREW

Minimum flight crew consists of one pilot who shall operate helicopter from right crew seat.

Left crew seat may be used for an additional pilot when approved dual controls are installed.

1-5. CONFIGURATION

1-5-A. REQUIRED EQUIPMENT

A functional flashlight is required for night flights.

FADEC system software shall be version 5.1.

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1-5-B. OPTIONAL EQUIPMENT

The snow deflector kit (BHT-407-FMS-4) shall be installed when conducting flight operations in falling and/or blowing snow.

Refer to appropriate flight manual supplement(s) (FMS) for additional limitations, procedures, and performance data for optional equipment.

1-5-C. DOORS REMOVED

NOTE

Indicated altitude may be up to 100 feet lower than actual altitude with crew door(s) removed.

Flight with any combination of doors removed is approved. With litter door removed, left passenger door shall be removed. Refer to Airspeed limitations.

With door(s) removed, determine weight change and adjust ballast if necessary. Refer to Section 5.

NOTE

All unsecured items shall be removed from cabin when any door is removed.

1-6. WEIGHT AND CENTER OF GRAVITY

1-6-A. WEIGHT

Maximum approved internal GW for takeoff and landing is 5000 pounds (2268 kilograms).

Minimum GW for flight is 2650 pounds (1202 kilograms).

Minimum weight at fuselage station 65.0 is 170 pounds (77.1 kilograms).

Maximum approved GW with jettisonable external load for takeoff and landings is 6000 pounds (2722 kilograms).

1-6-B. CENTER OF GRAVITY

The pilot is responsible for determining weight and balance to ensure gross weight and center of gravity will remain within limits throughout each flight. Refer to Section 5 for loading tables and instructions.

NOTE

Ballast as required to maintain most forward or most aft CG within GW flight limits (figure 1-1). For standard passenger and fuel loadings, applicable Weight empty center of gravity chart in BHT-407-MM-1 may be used to determine required ballast.

For longitudinal CG limits refer to Gross weight longitudinal center of gravity limits chart (figure 1-1).

For lateral CG limits refer to Gross weight lateral center of gravity limits (figure 1-2).

1-7. AIRSPEED

Helicopters S/N 53000 - 53074 basic V_{NE} is 130 KIAS, sea level to 3000 feet H_D. Decrease V_{NE} for ambient conditions in accordance with AIRSPEED LIMITATIONS placards and decals (figure 1-3).

Helicopters S/N 53000 - 53074 after compliance with Technical Bulletin 407-96-2, and S/N 53075 and Sub, increase basic V_{NE} to 140 KIAS, sea level to 3000 feet H_D. Decrease V_{NE} for ambient conditions in accordance with AIRSPEED LIMITATIONS placards and decals (figure 1-3).

V_{NE} at 93.5 to 100% TORQUE (takeoff power) is 100 KIAS, not to exceed placarded V_{NE}.

V_{NE} is 100 KIAS or placarded V_{NE}, whichever is less, when takeoff loading is in shaded area of the Gross weight lateral center of gravity limits (figure 1-2).

V_{NE} is 100 KIAS with any door(s) removed, not to exceed placarded V_{NE}.

V_{NE} is 100 KIAS or placarded V_{NE}, whichever is less for steady state autorotation.

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Maximum allowable airspeed for sideward and rearward flight or crosswind hover is 35 KTAS.

1-8. ALTITUDE

Maximum operating altitude is 20,000 feet H_p.

1-9. MANEUVERING

1-9-A. PROHIBITED MANEUVERS

Aerobatic maneuvers are prohibited.

1-9-B. CLIMB AND DESCENT

Maximum rate of climb is 2,000 feet per minute.

1-9-C. SLOPE LANDING

Slope landings are limited to 10° side slopes, 10° nose up slope or 5° nose down slope.

1-10. NOT USED

1-11. AMBIENT TEMPERATURES

Maximum sea level ambient air temperature for operation is +51.7°C (+125°F) and decreases with H_p at standard lapse rate of 2°C (3.6°F) per 1000 feet to 20,000 feet. Refer to Ambient air temperature limitations chart (Figure 1-4).

Minimum ambient air temperature for operation at all altitudes is –40 °C (–40°F).

ENG ANTI ICE shall be ON in visible moisture when OAT is below 5°C (40 °F).

1-12. ELECTRICAL

1-12-A. GENERATOR

Continuous operation, up to 10,000 feet H_p	0 to 180 Amps
Maximum continuous up to 10,000 feet H_p	180 Amps
Continuous operation, above 10,000 feet H_p	0 to 170 Amps
Maximum continuous above 10,000 feet H_p	170 Amps
Transient, 2 minutes	180 to 300 Amps
Transient, 5 seconds	300 to 400 Amps

1-12-B. STARTER

External Power Start	Battery Start
40 seconds ON	60 seconds ON
30 seconds OFF	60 seconds OFF
40 seconds ON	60 seconds ON
30 seconds OFF	60 seconds OFF
40 seconds ON	60 seconds ON
30 minutes OFF	30 minutes OFF

NOTE

28 VDC GPU for starting shall be limited to 500 amps.

1-13. POWER PLANT

Allison model 250-C47B.

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NOTE

Intentional use of any power transient is prohibited.

1-13-A. GAS PRODUCER RPM (NG)

Continuous operation	63 to 105%
Maximum continuous operation	105%
Transient, 10 seconds	105.1 to 106%

1-13-B. POWER TURBINE RPM (NP)

Minimum	99%
Continuous operation	99 to 100%
Maximum continuous	100%
Maximum transient, 15 seconds	104.3% NP at 100% TORQUE to 113.3% NP at 0% TORQUE

NOTE

When operating in MANUAL mode NP should be maintained between 95% and 100%.

1-13-C. MEASURED GAS TEMPERATURE (MGT)

GAGE P/N 407-375-001-101/-103

Continuous operation	100 to 727°C
Maximum continuous	727°C
Takeoff, 5 minutes	727 to 779°C
Maximum for takeoff	779°C
Transient, 12 seconds	780 to 826°C
Maximum starting, do not exceed 10 seconds above 826°C or 1 second at 927°C.	927 °C

NOTE

Either MGT gage may be installed.

GAGE P/N 407-375-001-105

Continuous operation	100 to 727°C
Maximum continuous	727°C
Takeoff, 5 minutes	727 to 779°C
Maximum for takeoff	779°C
Transient, 12 seconds	780 to 905°C
Maximum starting, do not exceed 10 seconds above 843°C or 1 second at 927°C.	927°C

1-13-D. ENGINE TORQUE

Continuous operation	0 to 93.5%
Maximum continuous	93.5%
Takeoff, 5 minute	93.5 to 100%
Transient, 5 seconds	105%

NOTE

Use of takeoff power is limited to 100 KIAS, not to exceed placarded V_{NE}.

1-13-E. FUEL PRESSURE

Minimum	8 PSI
Continuous operation	8 to 25 PSI
Maximum	25 PSI

1-13-F. ENGINE OIL PRESSURE

Minimum below 79% NG	50 PSI
Minimum from 79 to 94% NG	90 PSI
Minimum above 94% NG	115 PSI
Maximum	130 PSI
Maximum cold starts only	200 PSI

NOTE

When 130 PSI is exceeded during start, operate engine at idle until oil pressure drops below 130 PSI.

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1-13-G. ENGINE OIL TEMPERATURE

Continuous operation	0 to 107°C
Maximum	107°C

NOTE

Positive temperature indication is when the second segment of the trend arc is illuminated.

1-14. TRANSMISSION

1-14-A. TRANSMISSION OIL PRESSURE

Minimum	30 PSI
Continuous operation	40 to 70 PSI
Maximum	70 PSI

1-14-B. TRANSMISSION OIL TEMPERATURE

Continuous operation	15 to 110°C
Maximum	110°C

1-15. ROTOR

1-15-A. ROTOR RPM — POWER ON

Continuous operation	99 to 100%
Maximum continuous	100%

NOTE

When operating in MANUAL mode NR should be maintained between 95% and 100%.

1-15-B. ROTOR RPM — POWER OFF

Minimum	85%
Continuous operation	85 to 107%
Maximum	107%

CAUTION

FOR AUTOROTATIVE TRAINING MAINTAIN STEADY STATE NR ABOVE 90%.

1-16. HYDRAULIC

Hydraulic fluid MIL-H-5606 may be used at all ambient temperatures.

1-17. FUEL AND OIL

1-17-A. FUEL

Fuel conforming to following specifications may be used at all ambient temperatures:

ASTM-D-1655, Type B

MIL-T-5624, Grade JP-4 (NATO F-40).

Fuels conforming to following specifications are limited to ambient temperatures of −32°C (−25°F) and above:

ASTM-D-1655, Type A or A-1

MIL-T-5624, Grade JP-5 (NATO F-44)

MIL-T-83133, Grade JP-8 (NATO F-34).

For operations below −32 °C (−25 °F), refer to Allison Operation and Maintenance Manual for cold weather fuel and blending instructions.

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1-17-B. OIL

1-17-B-1. OIL – ENGINE

Oil conforming to MIL-L-7808 (NATO O-148), DOD-L-85734 (Turbine oil 555) or MIL-L-23699 (NATO O-156) is limited to ambient temperatures above –40 °C (–40 °F).

NOTE

Refer to Allison Operation and Maintenance Manual and BHT-407-MD-1 manual for approved oils and mixing of oils of different brands, types, and manufacturers.

1-17-B-2. OIL – TRANSMISSION AND TAIL ROTOR GEARBOX

NOTE

It is recommended DOD-L-85734 oil be used in transmission and tail rotor gearbox to maximum extent allowed by temperature limitations.

Oil conforming to DOD-L-85734 is limited to ambient temperatures above –40°C (–40°F).

Oil conforming to MIL-L-7808 (NATO O-148) is limited to ambient temperatures below –18°C (–0°F).

1-18. ROTOR BRAKE

Rotor brake (if installed) application is limited to ground operation after engine has been shut down and NR has decreased to 40% or lower.

For emergency stops, apply rotor brake any time after engine is shut down.

Engine starts with rotor brake engaged are prohibited.

1-19. NOT USED

1-20. INSTRUMENT MARKINGS AND PLACARDS

Refer to Figure 1-3 for Placards and decals. Refer to Figure 1-5 for Instrument markings.

NOTE

Illustrations shown in Figure 1-5 are artist representations and may or may not depict actual approved instruments due to printing limitations. Instrument operating ranges and limits shall agree with those presented in this section.

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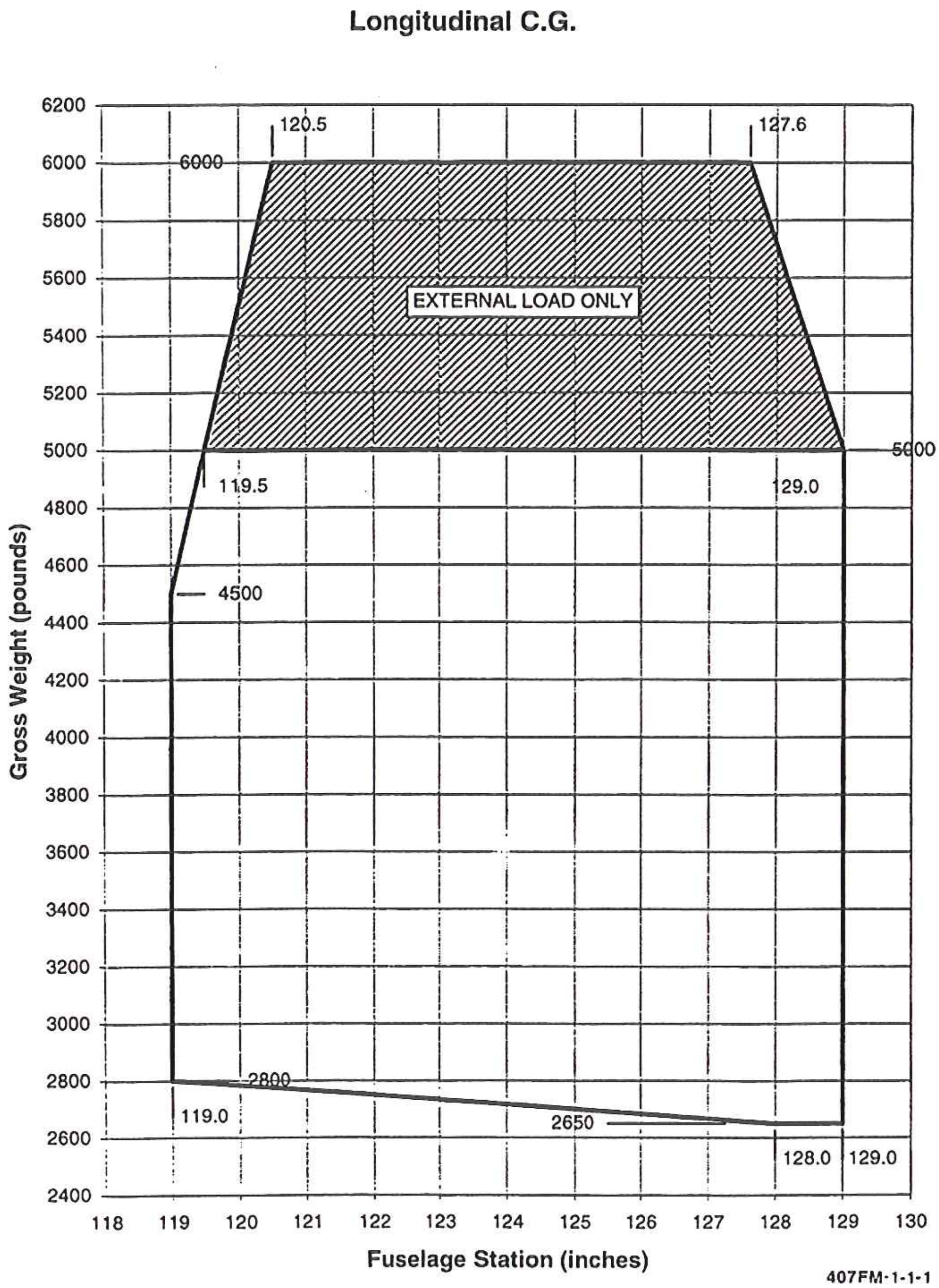

Figure 1-1. Gross weight longitudinal center of gravity limits (Sheet 1 of 2)

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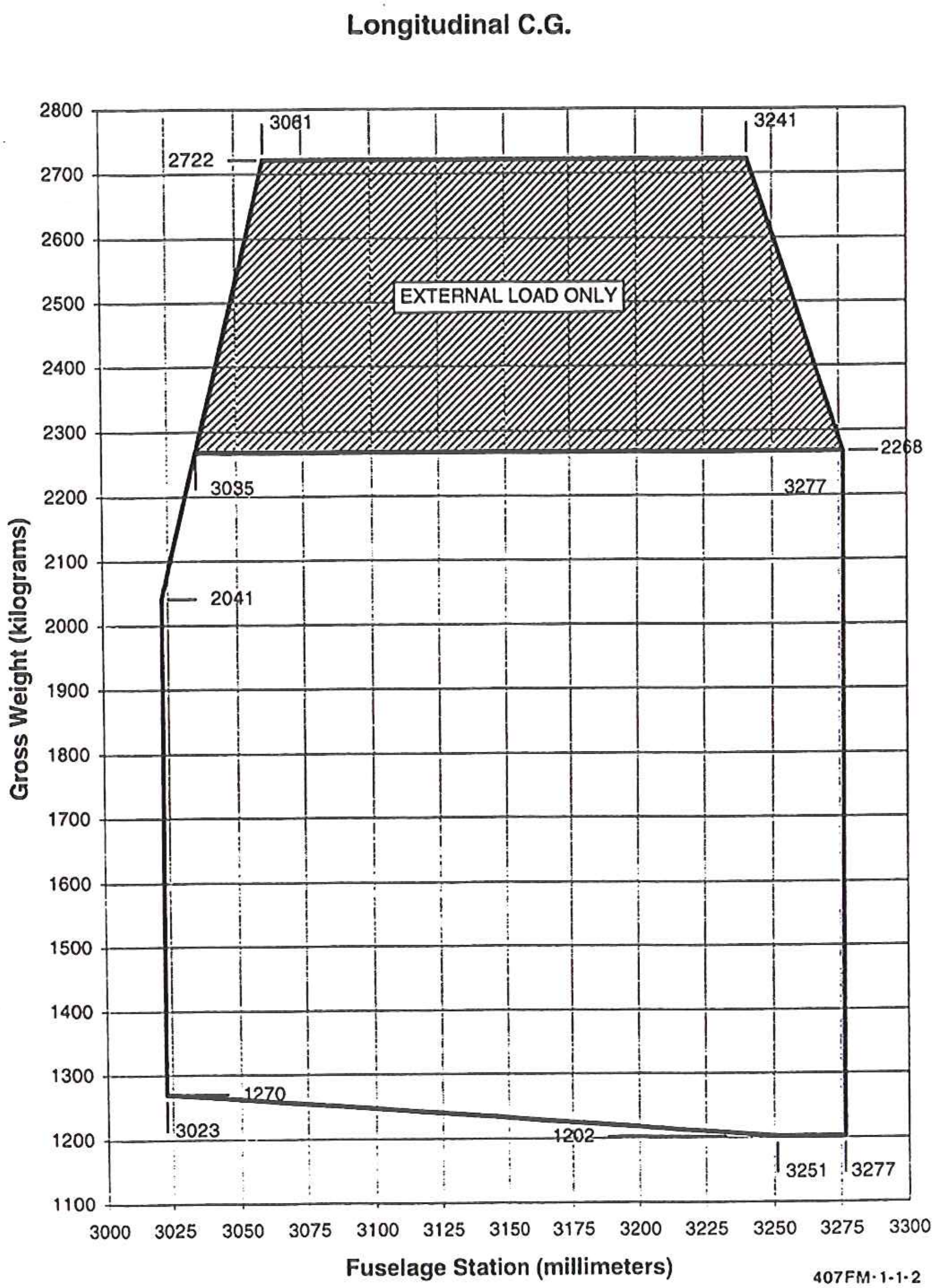

Figure 1-1. Gross weight longitudinal center of gravity limits (Sheet 2 of 2)

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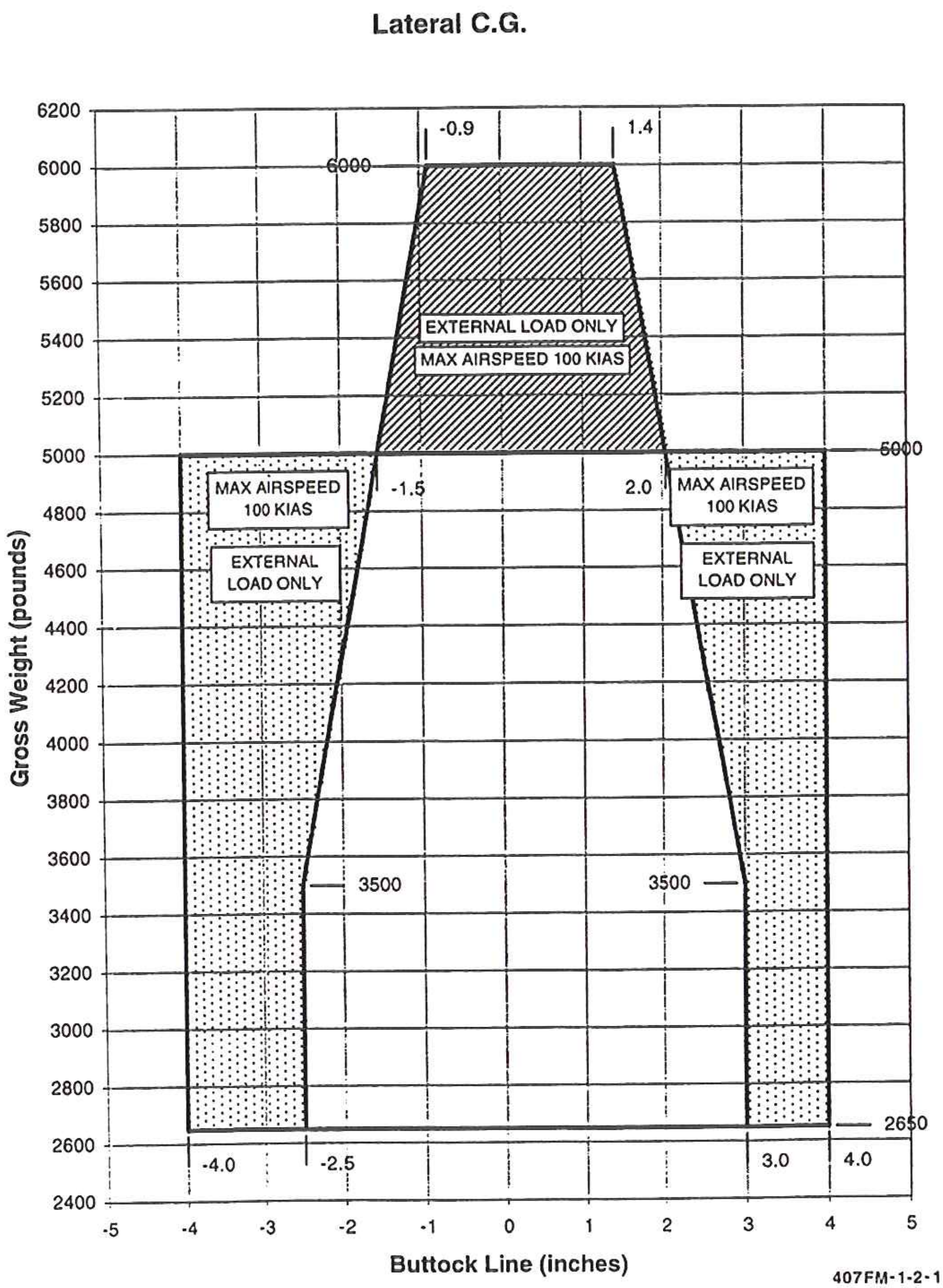

Figure 1-2. Gross weight lateral center of gravity limits (Sheet 1 of 2)

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Lateral C.G.

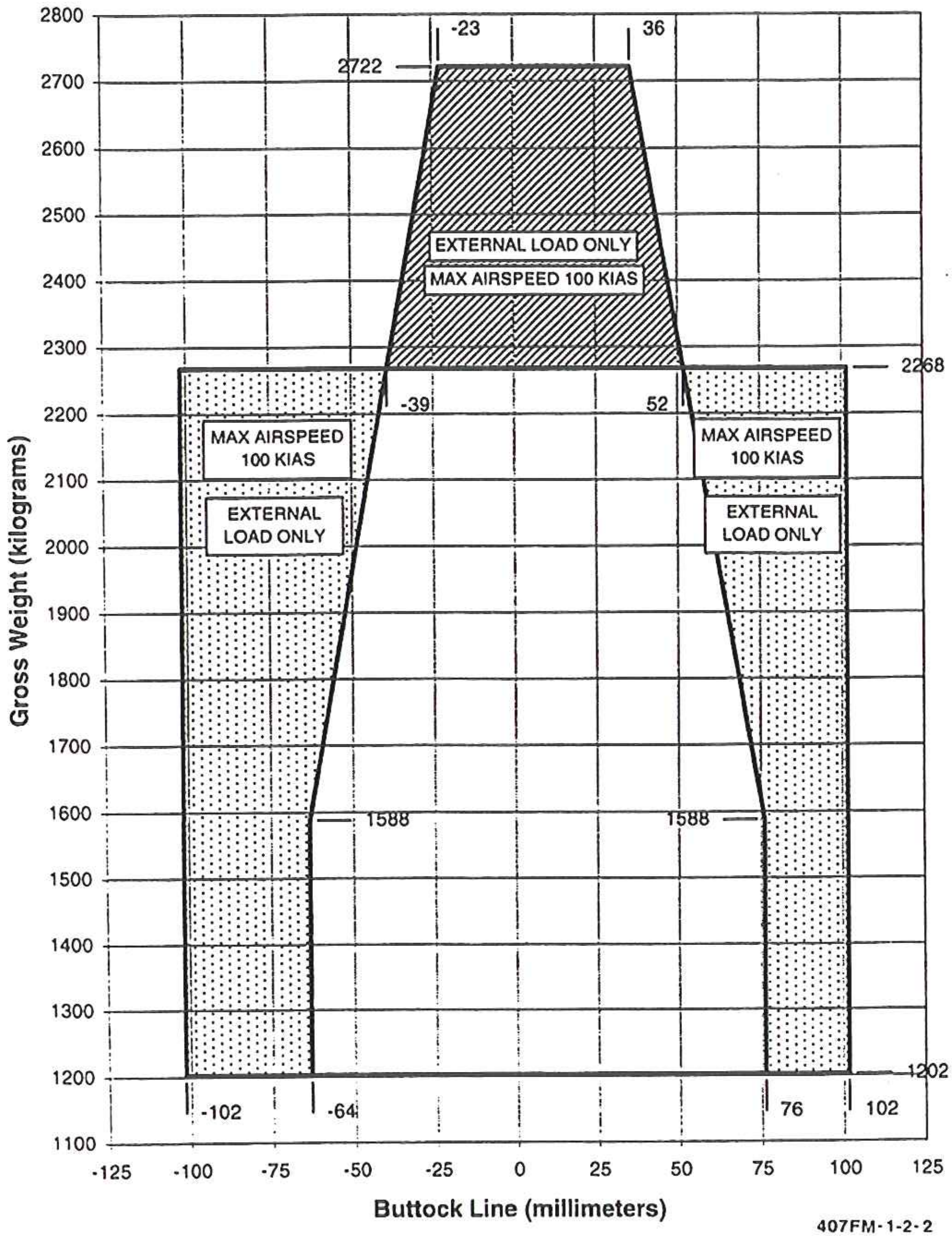

Figure 1-2. Gross weight lateral center of gravity limits (Sheet 2 of 2)

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407 AIRSPEED LIMITATIONS -KNOTS- IAS											
OAT	PRESSURE ALTITUDE FT x 1000										
C°	0	2	4	6	8	10	12	14	16	18	20
52	130	130	128	—	—	—	—	—	—	—	—
40	130	130	130	126	121	116	—	—	—	—	—
20	130	130	130	130	125	120	116	111	106	101	—
0	130	130	130	130	130	125	120	115	110	105	101
-20	122	122	122	122	122	122	122	120	116	110	105
-40	114	114	114	114	114	114	110	105	101	97	93
MAXIMUM AUTOROTATION VNE 100 KIAS											

S/N 53000-53074
LOCATION: FORWARD OF OVERHEAD CONSOLE

407 AIRSPEED LIMITATIONS -KNOTS- IAS											
OAT	PRESSURE ALTITUDE FT x 1000										
C°	0	2	4	6	8	10	12	14	16	18	20
52	138	133	128	—	—	—	—	—	—	—	—
40	140	135	131	126	121	116	—	—	—	—	—
20	140	140	135	130	125	120	116	111	106	101	—
0	140	140	140	135	130	125	120	115	110	105	100
-20	140	140	140	140	135	130	125	120	115	110	105
-40	137	133	128	123	119	114	110	105	101	97	93
MAXIMUM AUTOROTATION VNE 100 KIAS											

S/N 53075 and Sub or after compliance with TB 407-96-2
LOCATION: FORWARD OF OVERHEAD CONSOLE

407-FM-1-3-1

Figure 1-3. Placards and decals (Sheet 1 of 3)

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FUEL FUEL SYSTEM USABLE CAPACITY BASIC AIRCRAFT 127 U.S. GALLONS - 483 LITERS WITH 407-706-011 AUX KIT 147 U.S. GALLONS = 559 LITERS SEE FLIGHT MANUAL FOR APPROVED FUELS

Location: Above fuel filler cap.

THIS HELICOPTER MUST BE OPERATED IN COMPLIANCE WITH THE OPERATING LIMITATIONS SPECIFIED IN THE APPROVED FLIGHT MANUAL

Location: Bottom and centered on Instrument panel.

DO NOT APPLY ROTOR BRAKE ABOVE 40% RPM

Location: Near rotor brake (if installed).

CARGO MUST BE SECURED IN ACCORDANCE WITH FLIGHT MANUAL INSTR

Location: Inside of baggage door.

407-FM-1-3-2

Figure 1-3. Placards and decals (Sheet 2 of 3)

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FADEC SOFTWARE VERSION
5.1 INSTALLED. REFER TO
FLIGHT MANUAL FOR OPERATION.

Location: Instrument panel (with FADEC s/w V 5.1)

MAX ALLOWABLE WEIGHT 250 LBS.
MAX ALLOWABLE WEIGHT PER SQ. FT. 86 LBS.

Location: Inside of baggage door.

FUEL CAPACITY
BASIC 869 LBS
WITH AUX 1005 LBS
(JET A AT 15°C)

Location: Instrument panel

Location: Instrument panel and passenger compartment.

407-FM-1-3-3

Figure 1-3. Placards and decals (Sheet 3 of 3)

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INSTRUCTIONS FOR CHART USE

Enter chart at Pressure Altitude and proceed horizontally until altitude line intersects flight limit line. Proceed downward to read minimum and maximum ambient air temperature limits.

EXAMPLE:

Pressure Altitude is 6,000 feet.
What are minimum and maximum OAT values allowed for flight?
Answer: -40°C (-40°F) Minimum OAT
40°C (104°F) Maximum OAT

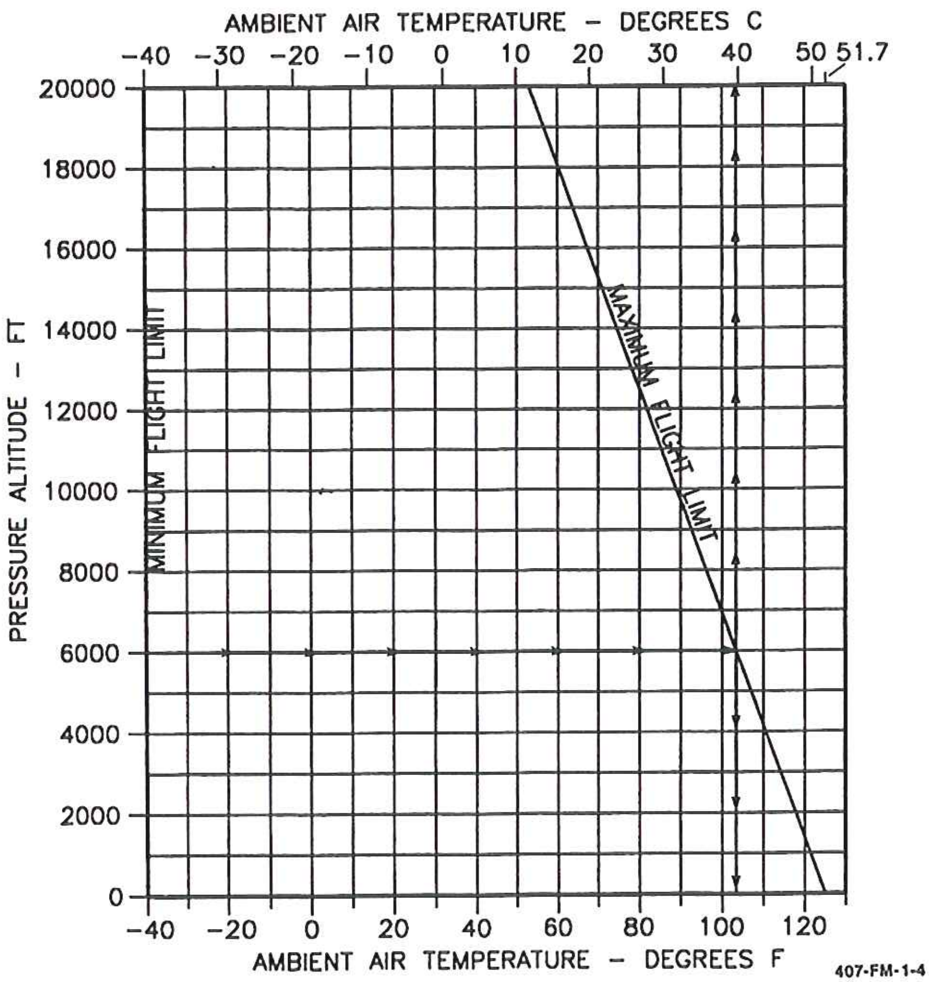

Figure 1-4. Ambient air temperature limitations

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ENGINE OIL PRESSURE

50 PSI	Minimum
50 to 90 PSI	Operation below 79% NG RPM
90 to 115 PSI	Continuous operation below 94% NG RPM
115 to 130 PSI	Continuous operation
130 PSI	Maximum for continuous operation
200 PSI	Maximum for cold start

ENGINE OIL TEMPERATURE

0 to 107°C	Continuous operation
107°C	Maximum

TRANSMISSION OIL PRESSURE

30 PSI	Minimum
40 to 70 PSI	Continuous operation
70 PSI	Maximum

TRANSMISSION OIL TEMPERATURE

15 to 110°C	Continuous operation
110°C	Maximum

NG (GAS PRODUCER RPM)

63 to 105%	Continuous operation
105%	Maximum continuous operation
106%	Maximum transient, 10 seconds

407-FM-1-5-1

Figure 1-5. Instrument markings (Sheet 1 of 4)

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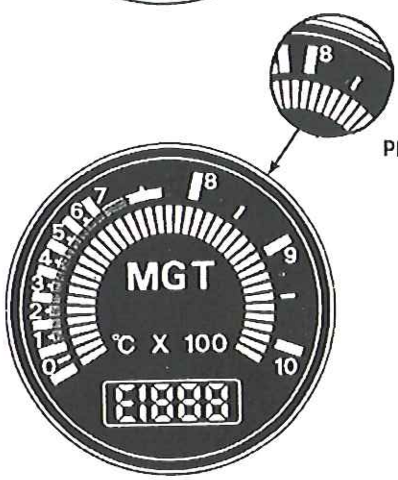
TRQ (TORQUE)

0 to 93.5%	Continuous operation
93.5 to 100%	5 minute takeoff range
100%	Maximum

* Either gage may be installed

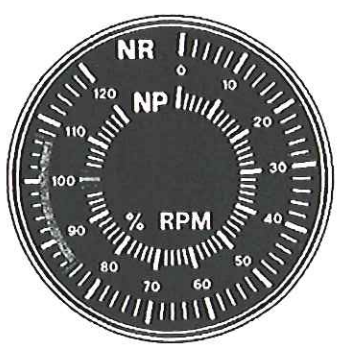


PN 407-375-001-105

MGT (MEASURED GAS TEMPERATURE)

100 to 727°C	Continuous operation
727 to 779°C	5 minute takeoff range
779°C	Maximum for takeoff
* 826°C or 843°C	Beginning of 10 seconds range for starting
927°C	Maximum for start and shutdown (1 second maximum)

PN 407-375-001-101/-103



NP (POWER TURBINE RPM)

99%	Minimum
99 TO 100%	Continuous operation
100%	Maximum continuous

NR (ROTOR RPM)

85%	Minimum (power off)
85 to 107%	Continuous operation (power off)
107%	Maximum (power off)

407-FM-1-5-2

Figure 1-5. Instrument markings (Sheet 2 of 4)

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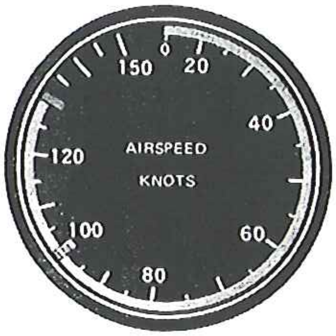


AIRSPEED

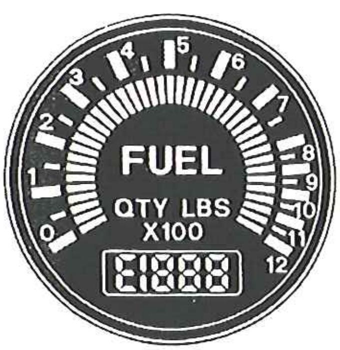
0 to 130 Knots	Continuous operation
100 Knots	Maximum for autorotation
130 Knots	Maximum

S/N 53000-53074

AIRSPEED

0 to 140 Knots	Continuous operation
100 Knots	Maximum for autorotation
140 Knots	Maximum

S/N 53075 and sub or after
compliance with TB 407-96-2



FUEL QUANTITY	(Jet A 6.8 lbs/gal)
0 LBS	All tanks empty (zero useable)
185 LBS	Forward tank empty
869 LBS	Forward and aft tanks full
1005 LBS	Forward, aft and auxiliary tanks full

407-FM-1-5-3

Figure 1-5. Instrument markings (Sheet 3 of 4)

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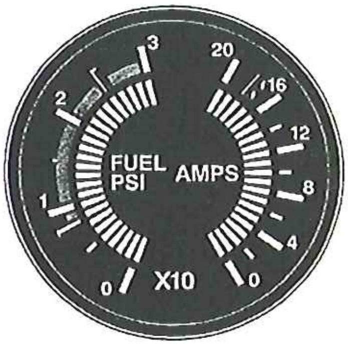
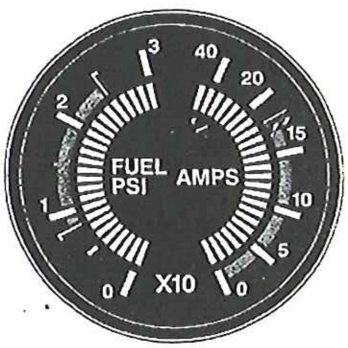
✳ Either gage may be Installed.

✳ P/N 407-075-024-101



DC LOAD		
170 Amps	Maximum continuous above 10,000 FT Hp	
180 Amps	Maximum	
FUEL PRESSURE		
8 PSI	Minimum	
8 to 25 PSI	Continuous operation	
25 PSI	Maximum	

✳ P/N 407-075-024-103
or 407-375-007-105



DC LOAD		
0 to 180 Amps	Continuous operation	
170 Amps	Maximum continuous above 10,000 FT Hp	
180 Amps	Maximum	
▲ 300 Amps	Maximum transient, 2 minutes	
● 400 Amps	Maximum transient, 5 seconds	
FUEL PRESSURE		
8 PSI	Minimum	
8 to 25 PSI	Continuous operation	
25 PSI	Maximum	

VERTICAL SPEED INDICATOR	
2,000 Feet Per minute up	Maximum

407-FM-1-5-4

Figure 1-5. Instrument markings (Sheet 4 of 4)

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Section 2

NORMAL PROCEDURES

2-1. INTRODUCTION

This section contains instructions and procedures for operating helicopter from planning stage, through actual flight conditions, to securing helicopter after landing.

Normal and standard conditions are assumed in these procedures. Pertinent data in other sections is referenced when applicable.

Instructions and procedures contained herein are written for purpose of standardization and are not applicable to all situations.

2-1-A. COLD WEATHER OPERATIONS

Battery starts have been demonstrated to -29°C (-20°F) with standard 17 amp-hour battery and -35°C (-31°F) with optional 28 amp-hour battery.

During engine start in cold temperatures initial engine oil pressure of 200 PSI and pressure excursions down to 50 PSI during warm up are normal. Normal oil pressure and temperature indications as per limitations section should be obtained after approximately 5 minutes at idle.

2-1-B. HOT WEATHER OPERATIONS

CAUTION

DURING EXTENDED HOVER AT TAKEOFF POWER WITH THE OAT ABOVE 49.7°C (121.4°F), MONITOR THE ENGINE OIL TEMPERATURE. IF TEMPERATURE RISES ABNORMALLY, REDUCE POWER OR TRANSITION TO FORWARD FLIGHT UNTIL TEMPERATURE DECREASES.

2-2. FLIGHT PLANNING

Each flight should be planned adequately to ensure safe operations and to provide pilot with data to be used during flight.

Check type of mission to be performed and destination.

Determine that aircraft has adequate performance to complete mission utilizing appropriate performance charts in Section 4.

Determine aircraft weight and balance will be within limits during entire mission. Utilize appropriate weight and balance charts in Section 5 and limitations in Section 1.

2-3. PREFLIGHT CHECK

Pilot is responsible for determining whether helicopter is in condition for a

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safe flight. Refer to Figure 2-1 for preflight check sequence.

NOTE

A preflight check is not intended to be a detailed mechanical inspection, but simply a guide to help pilot check condition of helicopter. It may be as comprehensive as conditions warrant at discretion of pilot.

All areas checked shall include a visual check for evidence of corrosion, particularly when helicopter is flown near salt water or in areas of high industrial emissions.

2-3-A. BEFORE EXTERIOR CHECK

1. Flight planning — Completed.
2. Publications — Checked.
3. GW and CG — Computed.
4. Helicopter servicing — Completed.
5. Battery — Connected.

2-3-B. EXTERIOR CHECK

2-3-B-1. FUSELAGE – CABIN RIGHT SIDE

WARNING

FAILURE TO REMOVE ROTOR TIEDOWNS BEFORE ENGINE STARTING MAY RESULT IN SEVERE DAMAGE AND POSSIBLE INJURY.

1. All main rotor blades — Tiedowns removed, condition.
2. Right static port — Condition.
3. Cabin doors and hinge bolts — Condition and security.
4. Windows — Condition and security.
5. Landing gear — Condition. Ground handling wheels removed.
6. Forward and aft crosstube fairings (if installed) — Secured, condition, and aligned.

2-3-B-2. FUSELAGE – CENTER RIGHT SIDE

1. Engine inlet — Condition; remove inlet covers.
2. Cabin roof, transmission cowling, and engine air inlet area — Cleaned of all debris, accumulated snow and ice; cowling secured.
3. Forward fairing — Secured.
4. Transmission — Check oil level. Verify actual presence of oil in sight gauge.
5. Transmission oil cooler lines — Condition and security.
6. Transmission mounts — Condition and security.
7. Main driveshaft — Condition.
8. Access door — Secured.
9. Fuel filler cap — Visually check fuel level and cap secured.

NOTE

If helicopter is not parked on a level surface fuel sump may not properly drain contaminants.

10. Fuel sump — Drain fuel sample as follows:

    a. RIGHT and LEFT FUEL BOOST/ XFR circuit breaker switches — OFF.

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b. BATT switch — BATT (on).

c. FUEL VALVE switch — OFF.

d. FWD and AFT FUEL SUMP drain buttons — Press, drain sample, then release.

11. Airframe fuel filter — Drain and check before first flight of day as follows:

a. RIGHT and LEFT FUEL BOOST/XFR circuit breaker switches — LEFT and RIGHT (on).

b. FUEL VALVE switch — ON.

c. Fuel filter drain valve — Open, drain sample, then close.

12. Fuel filter test switch — Press and check FUEL FILTER caution light illuminates. Release switch and check light extinguishes.

13. FUEL VALVE switch — OFF.

14. LEFT and RIGHT FUEL BOOST/XFR circuit breaker switches — OFF.

15. BATT switch — OFF.

16. Powerplant area:

a. Main driveshaft aft flexure — Condition.

b. Engine — Condition; security of attachments. Evidence of oil leakage.

c. Engine mounts — Condition and security.

d. Throttle linkage — Condition, security, and freedom of operation.

e. Engine fuel pump — Security and condition; evidence of leakage.

f. Hydromechanical unit — Security and condition; evidence of leakage.

g. Hoses and tubing — Chafing, security, and condition.

17. Engine cowl — Secured.

18. Generator cooling scoop — Clear of debris.

19. Oil tank — Leaks, security, and cap secured.

20. Access door — Secured.

21. Aft fairing — Secured.

2-3-B-3. FUSELAGE - AFT RIGHT SIDE

1. Fuselage — Condition.

2. Tail rotor driveshaft cover — Condition and security.

3. Tailboom — Condition.

4. Horizontal stabilizer and position light — Condition and security.

2-3-B-4. FUSELAGE - FULL AFT

1. Vertical fin — Condition.

2. Tail rotor guard — Condition and security.

3. Anticollision light — Condition and security of lens.

4. Aft position light — Condition.

5. Tail rotor gearbox — Oil level, leaks and security.

6. Tail rotor — Tiedown removed, condition and free movement.

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7. Tail rotor controls — Condition and security.

8. Tail rotor blades:

   a. General condition.

   b. Tip block — Security and seal integrity.

   c. Internal blade root — Clear of snow and ice.

9. Tail rotor yoke — Condition, evidence of static stop contact damage (deformed static stop yield indicator).

2-3-B-5. FUSELAGE – AFT LEFT SIDE

1. Tailboom — Condition.

2. Tail rotor driveshaft cover — Condition and security.

3. Horizontal stabilizer and position light — Condition and security.

4. Fuselage — Condition.

5. Forward tail rotor driveshaft coupling — Condition of splined adapter.

6. Oil cooler blower shaft hanger bearings — Evidence of grease leakage and overheating.

7. Oil cooler blower — Clear of obstructions and condition.

8. Oil cooler — Condition and leaks.

9. Oil cooler blower access door — Secured.

10. Oil tank sight glass — Check oil level.

11. Aft fairing — Secured.

12. Baggage compartment — Cargo tied down, door secured.

13. Exhaust cover — Removed.

14. Powerplant area:

    a. Engine — Condition, security of attachments.

    b. Engine mounts — Condition and security.

    c. Exhaust stack — Condition and security.

    d. Evidence of fuel and oil leaks.

    e. Fuel and oil filter bypass indicators — Check retracted.

    f. Hoses and tubing for chafing and condition.

    g. Pneumatic lines — Condition and security.

    h. Tail rotor driveshaft — Condition of splines and couplings.

    i. Air induction diffuser duct — Condition and security.

    j. Rotor brake disc and caliper (if installed) — Condition, security of attachment and leakage. Ensure brake pads are retracted from brake disc.

    k. Engine cowling — Secured.

    l. Air induction cowling — Secured.

    m. Cabin roof, transmission cowling, engine air inlet area, and plenum — Clear of all debris, accumulated snow and ice; cowling secured.

15. Transmission area:

    a. Transmission mounts — Condition and security of elastomeric mounts.

    b. Transmission oil filter — Bypass indicator retracted.

    c. Main driveshaft — Condition.

    d. Transducers and pressure lines — Condition and security.

    e. Access door — Secured.

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2-3-B-6. CABIN ROOF

1. Main rotor dampers and fairing — Condition and security.
2. Main rotor hub, yoke and frahm — Condition and security.
3. Main rotor blade and skin — Condition.
4. Pitch horn bearing — Wear and security.
5. Main rotor pitch links — Condition and security of attachment bolts and locking hardware.
6. Swashplate assembly — Condition, security of attached controls, and boot condition.
7. Control linkages to swashplate — Condition, security of attachment bolts and locking hardware.
8. Control tube hydraulics-off balance springs — Condition and security.
9. Hydraulic reservoir filler cap — Closed and locked.
10. Hydraulic system filters — Bypass indicator retracted.
11. Hydraulic actuators and lines — Condition, security, interference, leakage.

2-3-B-7. FUSELAGE – CABIN LEFT SIDE

1. Forward fairing and access door — Secured.
2. Cabin doors and hinge bolts — Condition and security.
3. Windows — Condition and security.
4. Hydraulic reservoir — Check fluid level.
5. Landing gear — Condition and ground handling wheels removed.
6. Forward and aft crosstube fairings (if installed) — Secured, condition, and aligned.
7. Left static port — Condition.

2-3-B-8. FUSELAGE – FRONT

1. Exterior surfaces — Condition.
2. Windshield — Condition and cleanliness.
3. Battery and vent lines — Condition and security.
4. HOUR METER circuit breaker — In.
5. Battery access door — Secured.
6. Pitot tube — Cover removed, clear of obstructions.
7. External power door — Condition and security.
8. Landing light lamps — Condition.
9. Antennas — Condition and security.

2-4. INTERIOR AND PRESTART CHECK

1. Cabin interior — Clean; equipment secured.
2. Fire extinguisher — Installed and secured.
3. Cabin loading — Maintain CG within limits.
4. Passenger seat belts — Secured.

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5. Copilot seat belt — Secured (if solo).

6. Doors — Secured.

7. Throttle — Closed.

8. LDG LTS switch — OFF.

9. Communications switches — Set.

10. Altimeter — Set.

11. Instruments — Correct indications.

12. Overhead switches — Set:

a. BATT switch — OFF.

b. GEN switch — OFF.

c. PART SEP switch (if installed) — OFF.

d. ANTI COLL LT switch — ANTI COLL LT (on).

e. HYD SYS switch — HYD SYS (on).

f. CABIN LT/PASS switch — OFF.

g. POS LT switch — As desired.

h. DEFOG switch — OFF.

i. PITOT HEATER switch — OFF.

j. ENG ANTI ICE switch — OFF.

k. AVIONICS MASTER switch — OFF.

l. HEATER switch (if installed) — OFF.

m. INSTR LT rheostat — OFF.

13. Overhead circuit breaker switches — OFF.

14. Overhead circuit breakers — In.

15. Rotor brake handle (if installed) — Up and latched.

CAUTION

28 VDC GPU SHALL BE 500 AMPERES OR LESS TO REDUCE RISK OF STARTER DAMAGE FROM OVERHEATING.

16. GPU — Connected (if used).

17. BATT switch — ON for battery start; ON for GPU start; OFF for battery cart start. Observe the following:

a. Low rotor audio horn activated.

b. For 8 seconds,

(1) Trend arcs on LCD instruments indicate full scale.

(2) TORQUE and NG digits display 8188.8.

(3) MGT and FUEL digits display 81888.

(4) NR and NP needles move to 107 and 100%, respectively.

c. After 3 seconds; ENG OUT, FADEC DEGRADE, FADEC FAULT, RESTART FAULT, and ENGINE OVSPD lights illuminate with activation of engine out audio for 3 seconds.

d. ENG OUT light re-illuminates with reactivation of engine out audio, after 3 seconds.

18. HORN MUTE button — Press to mute.

19. Caution lights — ENG OUT, XMSN OIL PRESS, RPM, HYDRAULIC SYSTEM, GEN FAIL, L/FUEL BOOST, R/FUEL BOOST, L/FUEL XFR, and R/FUEL XFR will be illuminated.

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NOTE

L/FUEL XFR and R/FUEL XFR will not be illuminated when forward fuel tank is empty.

20. Flight controls — Loosen frictions; check travel and verify CYCLIC CENTERING light operation; position for start. Tighten friction as desired.

21. Throttle — Check freedom of travel and idle stop. Return to closed position.

NOTE

With INSTR LT rheostat on and CAUT LT switch positioned to DIM, caution lights are dimmed to a fixed intensity and cannot be adjusted by INSTR LT rheostat.

22. INSTR LT rheostat — As desired.

23. CAUT LT switch — As desired.

24. FUEL BOOST/XFR circuit breaker switches — LEFT (on) and RIGHT (on) and verify all boost and transfer caution lights extinguish.

25. FUEL pressure — Check.

26. CAUTION LT TEST button — Press to test.

27. INSTR CHK button — Press and check for exceedances.

28. LCD TEST button — Press to test, if desired.

29. FADEC HORN TEST button — Press to test.

30. FADEC MODE switch — AUTO.

31. FUEL VALVE switch — ON, guard closed, FUEL VALVE light illuminates then extinguishes.

32. FUEL QTY — Check TOTAL and FWD tank quantity.

33. OAT/VOLTS display — Check OAT and select VOLTS.

CAUTION

ANY ATTEMPT TO START ENGINE WHEN VOLTAGE IS BELOW 24 VOLTS MAY RESULT IN A HOT START.

2-5. ENGINE START

1. Collective — Full down.

2. Cyclic and pedals — Centered and CYCLIC CENTERING light extinguished.

NOTE

If throttle is positioned in idle for more than 60 seconds, starter latching is disabled and throttle must be repositioned to cut off and then back to idle to enable it for another 60 seconds.

It is recommended that MGT be below 150°C when below 10,000 feet H_P or below 65 °C when above 10,000 feet H_P prior to attempting an engine start. Compliance with this recommendation will allow for cooler starts and reduce potential of reaching hot start abort limits. Refer to DRY MOTORING RUN, paragraph 2-5-A.

3. Throttle — Idle position.

4. START switch — Momentarily press (hold for approximately 1 second) and observe START and AUTO RELIGHT lights are illuminated.

5. MGT — Monitor.

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CAUTION

IF MAIN ROTOR IS NOT ROTATING BY 25% NG, ABORT START BY ROLLING THROTTLE TO CUT OFF. ENSURE STARTER HAS DISENGAGED WHEN MGT DECREASES BELOW 150°C.

6. START light — Extinguished at 50% NG (starter has disengaged).

7. AUTO RELIGHT light — Extinguished at 60% NG.

8. ENG and XMSN OIL pressures — Check.

CAUTION

IF ENGINE HAS BEEN SHUT DOWN FOR MORE THAN 15 MINUTES, STABILIZE AT IDLE FOR 1 MINUTE BEFORE INCREASING THROTTLE.

NOTE

During cold temperature operations, normal transmission and engine oil pressure limits may be exceeded during start. Stabilize engine at idle until minimum temperature and pressure limits are attained.

9. Idle — 63 ± 1% NG.

10. BATT switch — ON (if applicable).

11. GPU — Disconnect and close door (if applicable).

12. GEN switch — GEN (on); observe GEN FAIL light extinguishes.

NOTE

Turn generator OFF if ammeter indication drops to zero amps after an initial full scale indication. One reset is allowed. RESET generator and then turn generator back ON (applicable with AMPS/FUEL PSI gauge PN 407-075-024-101). Refer to BHT-407-MD-1.

13. Voltmeter — 28.5 ± 0.5 volts.

14. FLIGHT INSTR circuit breaker switches (3) (if installed) — DG, ATT and TURN (on).

NOTE

If dual controls are installed, guard throttle to prevent inadvertent manipulation from co-pilot position.

2-5-A. DRY MOTORING RUN

The following procedure is used to reduce residual MGT to recommended levels for engine start.

1. Throttle — Closed position.

2. START switch — Hold engaged for 15 seconds, then release.

Follow ENGINE START procedure, paragraph 2-5, once 0% NG is indicated.

2-6. SYSTEMS CHECK

2-6-A. PRELIMINARY HYDRAULIC SYSTEMS CHECK

NOTE

Uncommanded control movement or motoring with hydraulic system off may indicate hydraulic system malfunction.

1. HYD SYS switch — OFF.

2. HYDRAULIC SYSTEM caution light — Illuminated.

3. HYD SYS switch — HYD SYS (on).

4. HYDRAULIC SYSTEM caution light — Extinguished.

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2-6-B. FADEC MANUAL CHECK

WARNING

AUTO TO MANUAL MODE TRANSITIONS WITH N_R/N_P AT 100% FLAT PITCH CAN RESULT IN RAPID N_R/N_P ACCELERATION IN APPROXIMATELY 7 SECONDS. TO AVOID POSSIBLE OVERSPEED CONDITION, PERFORM THE FOLLOWING CHECK AT IDLE (63% N_G).

1. Throttle — Idle (63% NG).
2. FADEC MODE switch — MAN.
3. FADEC MANUAL and AUTO RELIGHT lights — Illuminated.

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4. Throttle — Increase slowly to ensure engine responds, then return to idle.

5. FADEC MODE switch - AUTO.

6. FADEC MANUAL and AUTO RELIGHT lights — Extinguished.

2-6-C. ENGINE RUNUP

1. Throttle — Increase smoothly to full open position. Check RPM warning light extinguished at 95% NR.

2. NR and NP needles — Check matching and indicating 100%.

NOTE

Overhead circuit breakers highlighted with arrow graphic ◪; are powered through AVIONICS MASTER switch.

3. AVIONICS MASTER switch — AVIONICS MASTER (on).

4. ELT (if installed) — Check for inadvertent transmission.

5. Flight controls — Check freedom with minimum friction.

6. ENG ANTI ICE switch — ENG ANTI ICE (on); check for MGT increase and illumination of ENGINE ANTI-ICE light (if installed).

7. ENG ANTI ICE switch — OFF; check MGT returns to normal and ENGINE ANTI-ICE light (if installed) extinguishes; then ENG ANTI ICE (on) if required.

NOTE

If temperature is below 5°C (40°F) and visible moisture is present, ENG ANTI ICE shall be on.

8. PART SEP switch (if installed) — As required.

2-6-D. HYDRAULIC SYSTEMS CHECK

NOTE

Hydraulic systems check is to determine proper operation of hydraulic actuators for each flight control system. If abnormal forces, unequal forces, control binding, or motoring are encountered, it may be an indication of a malfunctioning flight control actuator.

1. Collective — Full down.

2. NR — 100% RPM.

3. HYD SYS switch — OFF.

4. HYDRAULIC SYSTEM caution light — Illuminated.

5. Cyclic — Centered.

6. Cyclic control — Check normal operation by moving cyclic forward and aft, then left and right (approximately 1 inch). Center cyclic.

7. Collective — Check normal operation by increasing collective slightly (1 to 2 inches). Repeat two to three times as required. Return to full down position.

8. Pedals — Check normal operation by displacing pedals slightly (1 inch).

9. HYD SYS switch — HYD SYS (on).

10. HYDRAULIC SYSTEM caution light — Extinguished.

11. Cyclic and collective friction — Set as desired.

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2-7. BEFORE TAKEOFF

1. ENG ANTI ICE switch — As required.
2. Light switches — As required.
3. INSTR LT rheostat — As desired.

NOTE

For night flight, it is recommended to point the map light at the flight instruments and set to a low intensity. Sufficient night lighting will be provided in the event of an instrument lighting failure.

4. Radio(s) — Check as required.
5. Flight controls — Position and adjust frictions for takeoff.

NOTE

If throttle is not maintained in full open position during normal flight operations, available engine power may be limited.

6. Throttle — Full open. Check 99 to 100% NR/NP.
7. Engine, transmission, and electrical instruments — Within limits.
8. Flight and navigation instruments — Check.
9. FUEL QTY — Note indication.
10. FUEL QTY FWD TANK button — Press, note fuel remaining in forward cell.

2-8. TAKEOFF

1. Rear facing seat headrests — Adjusted to proper position.

NOTE

During takeoffs disregard CYCLIC CENTERING light and position cyclic as required.

2. Collective — Increase to hover.
3. Directional control — As required to maintain desired heading.
4. Cyclic — Apply as required to accelerate smoothly.
5. Increase collective, up to 5% torque above hover power, to obtain desired rate of climb and airspeed. Once clear of the HV diagram shaded areas, adjust power and airspeed as desired.

2-9. IN-FLIGHT OPERATIONS

1. AIRSPEED — As desired (not to exceed V_{NE} at flight altitude).

CAUTION

AT HIGH POWER AND HIGH AIRSPEED, CYCLIC ONLY ACCELERATIONS AND MANEUVERING MAY SIGNIFICANTLY INCREASE MGT AND TORQUE WITH NO COLLECTIVE INPUT. THIS INCREASE IS MORE RAPID AT LOWER OAT.

2. ENG ANTI ICE and PITOT HEATER switches — ENG ANTI ICE and PITOT HEATER switches on in visible moisture when ambient temperature is at or below 5°C (40°F).
3. PITOT HEATER — confirm operation (increase ammeter load).

NOTE

When ENG ANTI ICE switch is in ENG ANTI ICE (on), MGT will increase. Monitor MGT when selecting ENG ANTI ICE at high power settings.

4. Altimeter — Within limits.

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5. FUEL QTY FWD TANK button — Press, note forward fuel tank indication.

NOTE

Full forward fuel tank quantity (approximately 256 pounds) will be indicated at approximately 770 pounds or greater total fuel. Fuel transfer will be complete at approximately 185 pounds total fuel.

2-10. DESCENT AND LANDING

NOTE

Large reductions in collective pitch at heavy GW may permit NR to increase independent of NP (needles split). Main rotor may be reengaged with a smooth increase in collective pitch.

1. Rear facing seat headrests — Adjusted to proper position.
2. Flight controls — Adjust friction as desired.
3. Throttle — Full open. Check 99 to 100% NP.
4. Flight path — As required for type of approach.
5. ENG ANTI ICE — As required.
6. LDG LTS switch — As desired.

NOTE

During run-on or slope landings disregard CYCLIC CENTERING light and position cyclic as required. After landing is completed and collective is full down, reposition cyclic so that CYCLIC CENTERING light is extinguished.

2-11. ENGINE SHUTDOWN

1. Collective — Full down.
2. Cyclic and pedals — Centered and CYCLIC CENTERING light extinguished.
3. Cyclic friction — Increase so that cyclic maintains centered position.
4. LDG LTS switch — OFF.
5. Throttle — Reduce to idle stop. Check RPM warning light illuminated and audio on at 95% NR.

NOTE

If dual controls are installed, guard throttle to prevent inadvertent manipulation from co-pilot position.

6. HORN MUTE button — Press to mute.
7. MGT — Stabilize at idle for 2 minutes.
8. ENG ANTI ICE switch — OFF.
9. FLIGHT INSTR circuit breakers switches (if installed) — OFF
10. FUEL BOOST/XFR LEFT circuit breaker switch — OFF.

NOTE

Left fuel boost and transfer pumps will continue to operate until either LEFT FUEL BOOST/XFR circuit breaker switch (highlighted with yellow border) or FUEL VALVE switch is positioned to OFF. These pumps operate directly from battery and will not be deactivated when BATT switch is OFF. Battery power will be depleted if both switches remain on.

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11. ELT (if installed) — Check for inadvertent transmission.

12. AVIONICS MASTER switch — OFF.

13. GEN switch — OFF.

14. OVSPD TEST button — If required; press, hold 1 second, and release.

NOTE

Overspeed shut down test should be accomplished on first engine shut down of the day. ENGINE OVSPD light will momentarily illuminate in addition to those lights that illuminate during a normal shut down.

15. IDLE REL switch — Press and hold.

CAUTION

POSITIONING THROTTLE OUT OF CUT-OFF DURING NG SPOOL DOWN MAY CAUSE POST ENGINE SHUTDOWN FIRE.

16. Throttle — Closed; check MGT and NG decreasing, ENGINE OUT warning light illuminated and audio on at 55 ± 1%.

17. HORN MUTE button — Press to mute.

CAUTION

DO NOT USE COLLECTIVE TO SLOW ROTOR DURING COAST DOWN.

CAUTION

AVOID RAPID ENGAGEMENT OF ROTOR BRAKE IF HELICOPTER IS ON ICE OR OTHER SLIPPERY OR LOOSE SURFACE TO PREVENT ROTATION OF HELICOPTER.

18. Rotor brake (if installed) — Apply full rotor brake at or below 40% NR. Return rotor brake handle to stowed position just prior to main rotor stopping.

19. FUEL VALVE switch — OFF.

20. Pilot — Remain on flight controls until rotor has come to a complete stop.

21. All overhead switches, except HYD SYS switch — OFF.

NOTE

Ensure engine rotation has completely stopped prior to positioning BATT switch to OFF.

22. BATT switch — OFF, with NG at 0%.

CAUTION

APPLICABLE MAINTENANCE ACTION MUST BE PERFORMED PRIOR TO FURTHER FLIGHT IF A FADEC LIGHT HAS ILLUMINATED DURING THE PREVIOUS FLIGHT OR ON ENGINE SHUTDOWN.

NOTE

If shutting down at, or refueling to, between approximately 185 to 210 pounds total fuel quantity, up to 18 pounds of fuel may remain in forward fuel cell as unusable.

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2-12. POSTFLIGHT CHECK

If any of following conditions exist:

- Thunderstorms are in local area or forecasted.
- Winds in excess of 35 knots or a gust spread of 15 knots exists or is forecasted.
- Helicopter is parked within 150 feet of hovering or taxiing aircraft that are in excess of basic GW of helicopter.
- Helicopter to be left unattended.

Perform following:

1. Install main rotor blade tiedowns.
2. Secure tail rotor loosely to tailboom with tiedown strap to prevent excessive flapping.
3. Install exhaust cover, engine inlet protective plugs and pitot cover.

NOTE

Refer to BHT-407-MD-1 for additional tiedown data.

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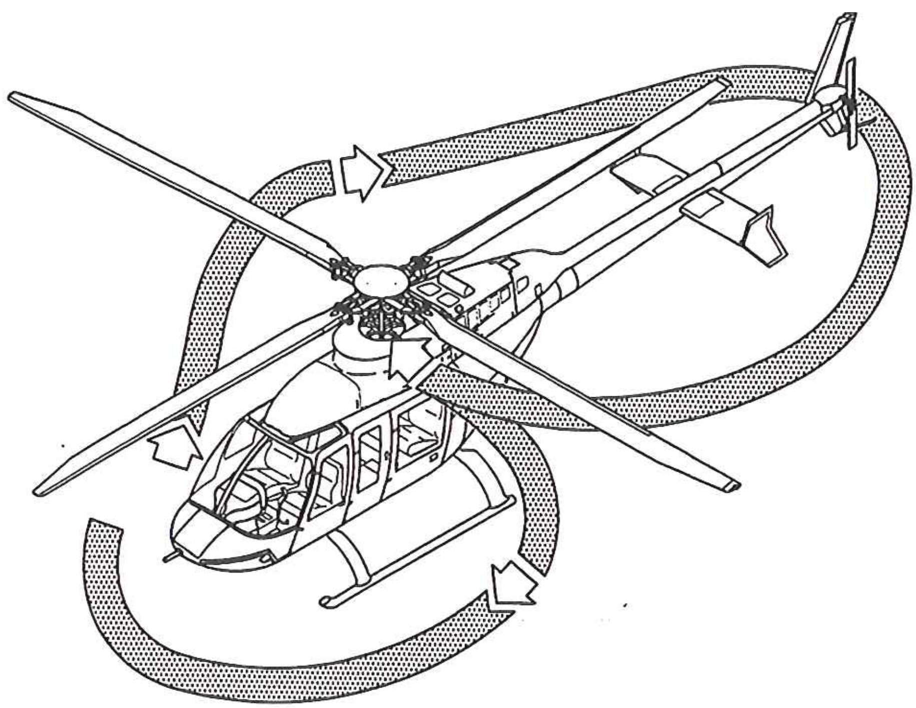

Figure 2-1. Preflight check sequence

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Section 3

EMERGENCY/MALFUNCTION PROCEDURES

3-1. INTRODUCTION

Following procedures contain indications of failures or malfunctions which affect safety of crew, helicopter, ground personnel or property; use of emergency features of primary and backup systems; and appropriate warnings, cautions, and explanatory notes. Tables 3-1 and 3-2 list fault conditions and corrective actions for warning lights and caution/advisory lights respectively.

Helicopter should not be operated following any precautionary landing until cause of malfunction has been determined and corrective maintenance action taken.

3-2. DEFINITIONS

Following terms indicate degree of urgency in landing helicopter.

LAND AS SOON AS POSSIBLE	Land without delay at nearest suitable area (i.e., open field) at which a safe approach and landing is reasonably assured.
LAND AS SOON AS PRACTICAL	Landing site and duration of flight are at discretion of pilot. Extended flight beyond nearest approved landing area is not recommended.

Following terms are used to describe operating condition of a system, subsystem, assembly, or component.

Affected	Fails to operate in intended or usual manner.
Normal	Operates in intended or usual manner.

All procedures listed herein assume pilot gives first priority to helicopter control and a safe flight path.

3-3. ENGINE

3-3-A. ENGINE FAILURE

3-3-A-1. ENGINE FAILURE — HOVERING

● INDICATIONS:

1. Left yaw.
2. ENGINE OUT and RPM warning lights illuminated.
3. Engine instruments indicate power loss.
4. Engine out audio activated when N_G drops below 55%.
5. N_R decreasing with RPM warning light and audio on when N_R drops below 95%.

● PROCEDURE:

1. Maintain heading and attitude control.

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2. Collective — Adjust to control NR and rate of descent. Increase prior to ground contact to cushion landing.

NOTE

Amplitude of collective movement is a function of height above ground. Any forward airspeed will aid in ability to cushion landing.

3. Land.
4. Shut down helicopter.

3-3-A-2. ENGINE FAILURE — INFLIGHT

● INDICATIONS:

1. Left yaw.
2. ENGINE OUT and RPM warning lights illuminated.
3. Engine instruments indicate power loss.
4. Engine out audio activated when NG drops below 55%.
5. NR decreasing with RPM warning light and audio on when NR drops below 95%.

● PROCEDURE:

1. Maintain heading and attitude control.
2. Collective — Adjust as required to maintain 85 to 107% NR.

NOTE

NR maintained at high end of operating range will provide maximum rotor energy to accomplish landing, but will cause an increased rate of descent.

3. Cyclic — Adjust to obtain desired autorotative AIRSPEED.

NOTE

Maximum AIRSPEED for steady state autorotation is 100 KIAS. Minimum rate of descent airspeed is 55 KIAS. Maximum glide distance airspeed is 80 KIAS.

4. Attempt engine restart if ample altitude remains. (Refer to ENGINE RESTART, paragraph 3-3-B).
5. FUEL VALVE switch — OFF.
6. At low altitude:
   a. Throttle — Closed.
   b. Flare to lose airspeed.
7. Apply collective as flare effect decreases to further reduce forward speed and cushion landing. Upon ground contact, collective shall be reduced smoothly while maintaining cyclic in neutral or centered position.
8. Complete helicopter shutdown.

3-3-B. ENGINE RESTART IN FLIGHT

NOTE

Engine restarts in flight have not been demonstrated.

An engine restart may be attempted in flight if time and altitude permit.

CAUTION

IF CAUSE OF FAILURE IS OBVIOUSLY MECHANICAL AS EVIDENCED BY ABNORMAL METALLIC OR GRINDING SOUNDS, DO NOT ATTEMPT A RESTART.

3-3-B-1. RESTART – AUTOMATIC MODE

● PROCEDURE (NO RESTART FAULT OR FADEC MANUAL LIGHTS ILLUMINATED):

1. Collective — Adjust to maintain 85 to 107% NR.
2. AIRSPEED — Adjust as desired.

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NOTE

Minimum rate of descent airspeed of 55 KIAS and minimum NR will allow pilot more time for air restart.

3. FUEL VALVE switch — ON.
4. Throttle — Cutoff; then reset to idle after NG is below 10%.
5. START switch — Engage.
6. MGT — Monitor.
7. Throttle — Advance smoothly to full open position.

If restart is unsuccessful, abort start and secure engine as follows:

8. Throttle — Closed.
9. FUEL VALVE switch — OFF.
10. Accomplish autorotative descent and landing.

3-3-B-2. RESTART — MANUAL MODE

RESTART FAULT OR FADEC MANUAL LIGHTS ILLUMINATED.

- PROCEDURE:

1. Collective — Adjust to maintain 85 to 107% NR.
2. AIRSPEED — Adjust as desired.

NOTE

Minimum rate of descent airspeed of 55 KIAS and minimum NR will allow pilot more time for air restart.

3. Throttle — Closed.
4. FADEC MODE switch — MAN.
5. FUEL VALVE switch — ON.
6. START switch — Hold to start position (starter will not latch).
7. NG — 12%.
8. Throttle — Slowly advance out of cutoff and stop advancing throttle at light off.
9. MGT — Allow to peak.
10. Throttle — Increase fuel flow by modulating throttle to maintain MGT within limits.
11. START switch — Release at 50% NG.
12. Throttle — Advance smoothly and modulate to 100% NP.

CAUTION

FOLLOWING A SUCCESSFUL MANUAL MODE START, DO NOT SWITCH TO AUTO MODE UNTIL AFTER LANDING. ENGINE FLAMEOUT MAY OCCUR. REFER TO ALLISON OPERATION AND MAINTENANCE MANUAL.

If restart is unsuccessful, abort start and secure engine as follows:

13. Throttle — Closed.
14. FUEL VALVE switch — OFF.
15. Accomplish autorotative descent and landing.

3-3-C. ENGINE UNDERSPEED

NO CAUTION/WARNING/ADVISORY LIGHTS ILLUMINATED.

- INDICATIONS:

1. Decrease in NG.
2. Subsequent decrease in NP.
3. Possible decrease in NR.
4. Decrease in TRQ.

- PROCEDURE:

1. Collective — Adjust as required to maintain 85 to 107% NR.
2. Throttle — Check open.

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3. Throttle — Position throttle to the approximate bezel position that coincides with the gage indicated NG.

4. FADEC MODE switch — MAN.

5. Throttle and collective — Adjust to maintain NR.

6. Land as soon as practical.

3-3-D. ENGINE OVERSPEED

(NO CAUTION/WARNING/ADVISORY LIGHTS ILLUMINATED)

● INDICATIONS:

1. Increase in NR.

2. Increase in NP.

3. Increase in NG.

4. Increase in TRQ.

● PROCEDURE:

1. Throttle — Retard.

2. NG or NP — Attempt to stabilize with throttle and collective.

3. FADEC MODE switch — MAN.

4. NR — Maintain with throttle and collective.

CAUTION

IF UNABLE TO MAINTAIN NR, NP, NG, OR MGT, PREPARE FOR A POWER OFF LANDING BY LOWERING COLLECTIVE AND SHUTTING DOWN ENGINE.

3-3-E. ENGINE COMPRESSOR STALL

● INDICATIONS:

1. Engine pops.

2. High or erratic MGT.

3. Decreasing or erratic NG or NP.

4. TRQ oscillations.

● PROCEDURE:

1. Collective — Reduce power, maintain slow cruise flight.

2. MGT and NG — Check for normal indications.

3. ENG ANTI ICE switch — ON.

4. PART SEP switch (if installed) — ON.

5. HEATER switch (if installed) — ON.

NOTE

Severity of compressor stalls will dictate if engine should be shut down and treated as an engine failure. Violent stalls can cause damage to engine and drive system components, and must be handled as an emergency condition. Stalls of a less severe nature (one or two low intensity pops) may permit continued operation of engine at a reduced power level, avoiding condition that resulted in compressor stall.

If pilot elects to continue flight:

6. Collective — Increase slowly to achieve desired power level.

7. MGT and NG — Monitor for normal response.

8. Land as soon as practical.

If pilot elects to shut down engine:

9. Enter autorotation.

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10. Throttle — Closed.
11. FUEL VALVE switch — OFF.
12. Collective — Adjust as required to maintain 85 to 107% NR.
13. Cyclic — Adjust as required to maintain desired AIRSPEED.
14. Prepare for power-off landing.

3-3-F. ENGINE HOT START/ SHUTDOWN

● INDICATIONS:

1. Excessive MGT.
2. Visible smoke or fire.

● PROCEDURE:

1. Throttle — Closed.
2. FUEL VALVE switch — OFF.

NOTE

Starter will remain engaged until MGT decreases to 150°C and then automatically disengage. Starter may be manually engaged by holding STARTER switch forward.

3. STARTER switch — Ensure starter is motoring engine until MGT stabilizes at normal temperature.
4. Shut down helicopter.

3-3-G. ENGINE OIL PRESSURE LOW OR FLUCTUATING

INDICATIONS:

1. Engine oil pressure below minimum.
2. Engine oil pressure fluctuating abnormally.

PROCEDURE:

1. Engine oil pressure and temperature — Monitor.
2. Land as soon as practical.

3-3-H. ENGINE OIL TEMPERATURE HIGH

INDICATIONS:

1. Engine oil temperature increasing above normal.
2. Engine oil temperature above maximum.

PROCEDURE:

Land as soon as practical.

3-3-J. DRIVESHAFT FAILURE

WARNING

FAILURE OF MAIN DRIVESHAFT TO TRANSMISSION WILL RESULT IN COMPLETE LOSS OF POWER TO MAIN ROTOR. ALTHOUGH COCKPIT INDICATIONS FOR A DRIVESHAFT FAILURE ARE SIMILAR TO AN ENGINE OVERSPEED, IT IS IMPERATIVE THAT AUTOROTATIVE FLIGHT PROCEDURES BE ESTABLISHED IMMEDIATELY. FAILURE TO REACT IMMEDIATELY TO LOW RPM AUDIO, RPM LIGHT AND NP/NR TACHOMETER CAN RESULT IN LOSS OF CONTROL.

INDICATIONS:

1. Left yaw
2. Rapid decrease in NR
3. Rapid increase in NP
4. LOW RPM audio horn
5. Illumination of RPM light
6. Possible increase in noise level due to overspeeding engine and driveshaft breakage.

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NOTE

Engine overspeed trip system will activate at 118.5% NP causing fuel flow to go to minimum. After initial overspeed, FADEC will adjust fuel flow to maintain engine at 100% N_P.

- PROCEDURE:

1. Maintain heading and attitude control.
2. Collective — Adjust as required to maintain 85 to 107% NR.

NOTE

Minimum rate of descent airspeed is 55 KIAS. Maximum glide distance airspeed is 80 KIAS.

3. Cyclic — Adjust to obtain desired autorotative airspeed.

NOTE

To maintain tail rotor effectiveness do not shutdown engine.

4. Landing — Complete autorotative landing.
5. Complete helicopter shutdown.

3-3-I. FADEC FAILURE

NOTE

Takeoff power may not be available in the MANUAL mode. Maximum continuous power will be available for all ambient conditions.

- INDICATIONS:

1. FADEC fail audio activated.
2. FADEC FAIL warning light illuminated.

- PROCEDURE:

WARNING

FAILURE TO FIXED FUEL FLOW - NP/NR WILL INCREASE RAPIDLY IF COLLECTIVE LOWERED, OR DECREASE IF RAISED.

WARNING

FAILURE TO MANUAL - WITHIN 2 TO 7 SECONDS AFTER THE FADEC FAIL WARNING, NR/NP MAY INCREASE VERY RAPIDLY, REQUIRING POSITIVE MOVEMENTS OF COLLECTIVE AND THROTTLE TO CONTROL NR.

NOTE

If time permits, matching throttle bezel position to NG indication will result in smooth transition to Manual Mode.

1. Throttle — Retard to approximately 90% throttle bezel position.
2. NR/NP — Control with collective only.
3. FADEC MODE switch — Depress one time.

NOTE

Initial engine response to manual control of fuel flow with throttle may take up to 7 seconds.

4. NR/NP — Maintain 95 to 100% with throttle and collective.
5. Land as soon as practical.
6. Normal shutdown if possible.

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3-4. FIRE

3-4-A. ENGINE FIRE ON GROUND

INDICATIONS:

1. Smoke
2. Fumes
3. Fire

PROCEDURES:

1. Throttle — Closed
2. FUEL VALVE switch — OFF
3. GEN switch — OFF
4. BATT switch — OFF
5. Rotor brake (if installed) — Engage
6. Exit helicopter

3-4-B. ENGINE FIRE DURING FLIGHT

● INDICATIONS:

1. Smoke.
2. Fumes.
3. Fire.

● PROCEDURE:

1. Inflight — Immediately enter autorotation.
2. Throttle — Closed.
3. FUEL VALVE switch — OFF.
4. If time permits, FUEL BOOST/XFR circuit breaker switches — OFF.
5. Execute autorotative descent and landing.
6. BATT switch — OFF.

NOTE

Do not restart engine until corrective maintenance has been performed.

3-4-C. CABIN SMOKE OR FUMES

INDICATIONS:

1. Smoke
2. Fumes

PROCEDURE:

1. Inflight — Start descent
2. AIR COND BLO switch (if installed) — OFF
3. HEATER switch (if installed) — OFF
4. All vents — Open
5. Side windows — Open

If time and altitude permits:

1. Source — Attempt to identify and secure
2. If source is identified and smoke and/or fumes still persist — Land as soon as possible.
3. If source is identified and smoke and/or fumes are cleared — Land as soon as practical.

3-5. TAIL ROTOR

There is no single emergency procedure for all types of antitorque malfunctions . One key to a pilot successfully handling a tail rotor emergency lies in the ability to quickly recognize the type of malfunction that has occurred.

3-5-A. COMPLETE LOSS OF TAIL ROTOR THRUST

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This is a situation involving a break in drive system (e.g., severed driveshaft), wherein tail rotor stops turning and delivers no thrust.

● INDICATIONS:

1. Uncontrollable yawing to right (left side slip).
2. Nose down tucking.
3. Possible roll of fuselage.

NOTE

Severity of initial reaction of helicopter will be affected by AIRSPEED, CG, power being used, and H_D.

● PROCEDURE:

3-5-A-1. HOVERING

Close throttle and perform a hovering autorotation landing. A slight rotation can be expected on touchdown.

3-5-A-2. IN-FLIGHT

Reduce throttle to idle, immediately enter autorotation, and maintain a minimum AIRSPEED of 55 KIAS during descent.

NOTE

When a suitable landing site is not available, vertical fin may permit controlled flight at low power levels and sufficient AIRSPEED. During final stages of approach, a mild flare should be executed, making sure all power to rotor is off. Maintain helicopter in a slight flare and smoothly use collective to execute a soft, slightly nose-high landing. Landing on aft portion of skids will tend to correct side drift. This technique will, in most cases, result in a run-on type landing.

3-5-B. FIXED PITCH FAILURES

This is a situation involving inability to change tail rotor thrust (blade angle) with anti-torque pedals.

● INDICATIONS:

1. Lack of directional response.
2. Locked pedals.

NOTE

If pedals cannot be moved with a moderate amount of force, do not attempt to apply a maximum effort, since a more serious malfunction could result. If helicopter is in a trimmed condition when malfunction occurs, TRQ and AIRSPEED should be noted and helicopter flown to a suitable landing area. Certain combinations of TRQ, NR, and AIRSPEED will correct a yaw attitude, and these combinations should be used to land helicopter.

● PROCEDURE:

3-5-B-1. HOVERING

Do not close throttle unless a severe right yaw occurs. If pedals lock in any position at a hover, landing from a hover can be accomplished with greater safety under power-controlled flight rather than by closing throttle and entering autorotation.

3-5-B-2. IN-FLIGHT — LEFT PEDAL APPLIED

In a high power condition, helicopter will yaw to left when power is reduced. Power and AIRSPEED should be adjusted to a value where a comfortable yaw angle can be maintained. If AIRSPEED is increased, vertical fin will become more effective and an increased left yaw attitude will develop. To accomplish landing, establish a power-on approach with sufficiently low AIRSPEED (zero if necessary) to attain a

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rate of descent with a comfortable sideslip angle. (A decrease in NP decreases tail rotor thrust.) As collective is increased just before touchdown, left yaw will be reduced.

3-5-B-3. IN-FLIGHT — RIGHT PEDAL APPLIED

In cruise flight or reduced power situation, helicopter will yaw to right when power is increased. A low power, run-on type landing will be necessary by gradually reducing throttle to maintain heading while adding collective to cushion landing. If right yaw becomes excessive, close throttle completely.

3-6. HYDRAULIC SYSTEM

3-6-A. LOSS OF HYDRAULIC PRESSURE

● INDICATIONS:

1. HYDRAULIC SYSTEM caution light illuminated.
2. Grinding or howling noise from pump.
3. Increase in force required to move flight controls.
4. Feedback forces may be evident during flight control movement.

● PROCEDURE:

1. Reduce AIRSPEED to 70 to 100 KIAS.
2. HYD SYSTEM circuit breaker — Out. If hydraulic power is not restored, push breaker in.
3. HYD SYS switch — HYD SYS; OFF if hydraulic power is not restored.
4. For extended flight set comfortable AIRSPEED, up to 120 KIAS, to minimize control forces.
5. Land as soon as practical.
6. A run-on landing at effective translational lift speed (approximately 15 knots) is recommended.

3-6-B. FLIGHT CONTROL ACTUATOR HARDOVER

An actuator hardover can occur in any flight control axis, but a cyclic cam jam will only occur in the fore and aft axis. An actuator hardover is manifested by uncommanded movements of one or two flight controls. If two controls move, the pilot will find one of these controls will require a higher than normal control force to oppose the movement. This force cannot be "trimmed" to zero without turning the HYD SYS switch OFF. Once the hydraulic boost is OFF, the forces on the affected flight control will be similar to the "normal" hydraulic off forces.

INDICATIONS:

1. Uncommanded flight control movements
2. High flight control forces to oppose movement in one axis
3. Feedback forces only in affected flight control axis
4. Flight control forces normal in unaffected axis

PROCEDURE:

1. Attitude — Maintain
2. HYD SYS switch — OFF
3. AIRSPEED — Set to 70 to 100 KIAS
4. Land as soon as possible using procedure from paragraph 3-6-A

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3-7. ELECTRICAL SYSTEM

3-7-A. GENERATOR FAILURE

● INDICATIONS:

1. GEN FAIL caution light illuminated.
2. AMPS indicates 0.
3. Voltmeter — Approximately 24 volts

● PROCEDURE:

1. GENERATOR FIELD and GENERATOR RESET circuit breakers — Check in.
2. GEN switch — RESET; then GEN.
3. If power is not restored, place GEN switch to OFF; land as soon as practical.

NOTE

With generator OFF, a fully charged battery will provide approximately 21 minutes of power for basic helicopter and one VHF COMM radio (35 minutes with optional 28 ampere/hour battery).

3-7-B. EXCESSIVE ELECTRICAL LOAD

● INDICATIONS:

1. AMPS indicates excessive load.
2. Smoke or fumes.

● PROCEDURE:

1. GEN switch — OFF.
2. BATT switch — OFF.
3. FUEL BOOST/XFR LEFT circuit breaker switch — LEFT (on).

WARNING

PRIOR TO BATTERY DEPLETION, ALTITUDE MUST BE REDUCED BELOW 8000 FEET H_P (Jet A) OR 4000 H_P (JET B). UNUSABLE FUEL MAY BE AS HIGH AS 135 POUNDS AFTER THE BATTERY IS DEPLETED DUE TO INABILITY TO TRANSFER FUEL FROM FORWARD CELLS.

NOTE

With battery and generator OFF, a fully charged battery will operate left fuel boost pump and left fuel transfer pump for approximately 2 hours (3.5 hours with optional 28 ampere/hour battery).

4. Land as soon as practical.

NOTE

When throttle is repositioned to the idle stop (during engine shutdown) the PMA will go offline and the engine may flameout.

3-8. FUEL SYSTEM

DUAL FUEL TRANSFER FAILURE

● INDICATIONS:

1. L/FUEL XFR and R/FUEL XFR caution lights illuminate.
2. Last 135 pounds of fuel in forward cell may not be usable.
3. Fuel will stop transferring from forward to aft fuel cell at approximately 340 pounds total indicated fuel.

● PROCEDURE:

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1. LEFT and RIGHT FUEL BOOST/XFR circuit breaker switches — Check ON.

2. Determine FUEL QTY in forward cell.

3. Subtract quantity of fuel trapped in forward cell from total to determine usable fuel remaining.

4. Plan landing accordingly.

3-9. AUTOMATIC FLIGHT CONTROLS SYSTEM

3-10. COMMUNICATION SYSTEM

3-11. CABIN HEATER

3-12. LANDING GEAR

3-13. LANDING GEAR CYCLIC CAM JAM

A cyclic cam jam can only occur in the fore and aft axis, whereas, an actuator hardover can occur in any flight control axis. A cyclic cam jam is manifested when a commanded control movement requires a higher than normal fore and aft spring force. The force felt when moving the cyclic fore and aft with a cam jam is the result of overriding a spring capsule.

- INDICATIONS:

  1. High (approximately 15 pounds) fore and aft cyclic control forces.

  2. Normal pedal, collective and lateral cyclic control forces.

- PROCEDURE:

  1. Helicopter pitch attitude — Maintain normal pitch attitudes with forward or aft cyclic force.

CAUTION

DO NOT TURN HYDRAULIC BOOST OFF

  2. Land as soon as practical.

3-15. WARNING, CAUTION, AND ADVISORY LIGHTS/MESSAGES

Red warning lights/messages, fault conditions, and corrective actions are presented in Table 3-1.

Amber caution and White advisory lights/ messages and corrective actions are presented in Table 3-2.

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Table 3-1. Warning (red) lights

PANEL WORDING	FAULT CONDITION	CORRECTIVE ACTION
BATTERY HOT	Battery overheating.	Turn BATT switch OFF and land as soon as practical. If BATTERY RLY light illuminates, turn GEN switch OFF if conditions permit. Land as soon as possible.
ENGINE OUT	NG less than 55 ± 1% and/or FADEC senses ENGINE OUT.	Verify engine condition. Accomplish engine failure procedure.
ENGINE OVSPD	NG greater than 110% or NP versus TORQUE is above maximum continuous limit (102.4% NP at 100% TORQUE to 108.6% NP at 0% TORQUE).	Adjust throttle and collective as necessary. Determine if engine is controllable, if not shut down. Maintenance action required before next flight.
FADEC FAIL(During start)	FADEC has detected a serious malfunction.	Close throttle immediately. Engage starter to reduce MGT. Applicable maintenance action required prior to next flight.
FADEC FAIL(Inflight)	FADEC has detected a malfunction and an overspeed may occur 2 to 7 seconds following activation of FADEC fail horn and illumination of FADEC FAIL warning light. Engine may underspeed significantly prior to overspeed. Any other FADEC related lights may be illuminated.	Accomplish FADEC FAILURE procedure, paragraph 3-3-K. Applicable maintenance action required prior to next flight.

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Table 3-1. Warning (red) lights (Cont)

PANEL WORDING	FAULT CONDITION	CORRECTIVE ACTION
RPM (with low RPM audio)	NR below 95%.	Reduce collective and ensure throttle is full open. Light will extinguish and audio will cease when NR increases above 95%.
RPM (without audio)	NR above 107%.	Increase collective and/or reduce severity of maneuver. Light will extinguish when NR decreases below 107%.
XMSN OIL PRESS	Transmission oil pressure is below minimum.	Reduce power; verify fault with gage. Land as soon as possible.
XMSN OIL TEMP	Transmission oil temperature is at or above red line.	Reduce power; verify fault with gage. Land as soon as practical.

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Table 3-2. Caution (amber) and advisory (white/green)lights

PANEL WORDING	FAULT CONDITION	CORRECTIVE ACTION
AUTO RELIGHT (white)	Engine igniter is operating.	None. NOTE AUTO RELIGHT light will be illuminated when ignition system is activated. Ignition system is activated: 1 - during start sequence 2 - in MANUAL mode with NG above 55% 3 - with FADEC detection of engine out condition with NG above 50%.
BAGGAGE DOOR	Baggage compartment door not securely latched.	Close door securely before flight. If light illuminates during flight, land as soon as practical.
BATTERY RLY	Battery relay has malfunctioned to closed (ON) position with BATT switch OFF. Battery is still connected to DC BUSS.	If BATTERY HOT light is illuminated, turn GEN switch OFF if conditions permit. Land as soon as possible.
CHECK INSTR	TRQ, MGT, or NG is about to or has detected an exceedance. Flashing LCD trend arc and digital display indicates impending exceedance. Letter E in digital display indicates an exceedance has occurred.	Reduce engine power if possible. Press INSTR CHK button to display magnitude of exceedance. Refer to BHT-407-MD-1.
CYCLIC CENTERING	Cyclic stick is not centered	Reposition cyclic stick to center position to extinguish CYCLIC CENTERING light.
ENGINE ANTI-ICE (white)	ANTI-ICE switch ON. Engine receiving anti-icing air.	If light (if installed) remains illuminated with ENGINE ANTI-ICE switch OFF, avoid operations requiring maximum power.
ENGINE CHIP	Ferrous particles in engine oil.	Land as soon as possible.

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Table 3-2. Caution (amber) and advisory (white/green)lights

PANEL WORDING	FAULT CONDITION	CORRECTIVE ACTION
FADEC DEGRADED(Inflight)	FADEC ECU operation is degraded which may result in NR droop, NR lag, or reduced maximum power capability.	Remain in AUTO mode. Fly helicopter smoothly and nonaggressively. Land as soon as practical. **NOTE** It may be necessary to use FUEL VALVE switch to shut down engine after landing. Applicable maintenance action required prior to next flight.
FADEC DEGRADED(With engine shutdown)	FADEC ECU has recorded a fault during previous flight or a current fault has been detected.	Position throttle to idle; if light extinguishes, fault is from previous flight. Applicable maintenance action required prior to next flight.
FADEC FAULT	MGT,NP or NG automatic limiting circuit(s) not functional.	Remain in AUTO mode. Land as soon as practical. Applicable maintenance action required prior to next flight.
FADEC MANUAL	FADEC is operating in MANUAL mode. No automatic governing is available. AUTO RELIGHT light will be illuminated.	Fly helicopter smoothly and nonaggressively. Maintain NR with coordinated throttle and collective movements. Land as soon as practical.
FLOAT ARM	FLOAT ARM switch is ON. Float inflation solenoid is armed.	Normal operation for takeoff and landing over water. FLOAT ARM switch — OFF. If light remains illuminated, FLOATS circuit breaker — Out. Land as soon as practical. **NOTE** With float inflation solenoid armed, flight should not exceed 60 KIAS and 500 feet AGL.
FLOAT TEST(green)	Float system in test mode.	None.
FUEL FILTER	Airframe fuel filter in impending bypass.	Land as soon as practical. Clean before next flight.
FUEL LOW	110 ±15 pounds of fuel remain in aft tank.	Verify FUEL QTY. Land as soon as practical.

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Table 3-2. Caution (amber) and advisory (white/green)lights (Cont)

PANEL WORDING	FAULT CONDITION	CORRECTIVE ACTION
R/FUEL BOOST	Right fuel boost pump has failed.	If practical, descend below 8000 feet H_P if fuel is Jet A or 4000 feet H_P if fuel is Jet B to prevent fuel starvation if other fuel boost pump fails or has low output pressure. Land as soon as practical. NOTE If either or both fuel boost pumps fail, unusable fuel is unaffected. Corrective action is as per R/FUEL BOOST or L/FUEL BOOST light.
L/FUEL BOOST	Left fuel boost pump has failed.	If practical, descend below 8000 feet H_P if fuel is Jet A or 4000 feet H_P if fuel is Jet B to prevent fuel starvation if other fuel boost pump fails or has low output pressure. Land as soon as practical.
FUEL VALVE	Fuel valve position differs from FUEL VALVE switch indication or FUEL VALVE circuit breaker out.	Check FUEL VALVE circuit breaker in. Land a soon as practical. If on ground, cycle FUEL VALVE switch.
L/FUEL XFR	Left fuel transfer pump has failed.	Land as soon as practical.

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Table 3-2. Caution (amber) and advisory (white/green)lights (Cont)

PANEL WORDING	FAULT CONDITION	CORRECTIVE ACTION
		CAUTION IF BOTH FUEL TRANSFER PUMPS FAIL, UNUSABLE FUEL MAY BE AS HIGH AS 135 POUNDS DUE TO INABILITY TO TRANSFER FUEL FROM FORWARD CELL. LAND AS SOON AS PRACTICAL.
	NOTE Under normal fuel transfer conditions, helicopters S/N 53000 - 53174 L/FUEL XFR and R/FUEL XFR lights will illuminate for 2 ½ minutes and then extinguish. This indicates transfer is complete and transfer pumps have been automatically turned off. Helicopters S/N 53175 and subsequent inhibit illumination of the lights.	
R/FUEL XFR	Right fuel transfer pump has failed.	Land as soon as practical.
GEN FAIL	Generator not connected to DC BUSS.	Verify fault with AMPS gage. GEN switch — RESET, then ON. If GEN FAIL light remains illuminated, GEN switch — OFF. Land as soon as practical.
HEATER OVERTEMP	An overtemp condition has been detected by a temperature probe either under pilot seat, copilot seat, or in vertical tunnel.	Turn HEATER switch OFF immediately.
HYDRAULIC SYSTEM	Hydraulic pressure below limit.	Verify HYD SYS switch position. Accomplish hydraulic system failure procedure (refer to paragraph 3-6).
LITTER DOOR	Litter door not securely latched.	Close door securely before flight. If light illuminates during flight, land as soon as practical.

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Table 3-2. Caution (amber) and advisory (white/green)lights

PANEL WORDING	FAULT CONDITION	CORRECTIVE ACTION
RESTART FAULT (white)	FADEC ECU has detected a fault which will not allow engine to be restarted in AUTO mode.	Remain in AUTO mode. Plan landing site accordingly. Applicable maintenance action required prior to next flight NOTE When throttle is repositioned to idle stop (during engine shutdown) the PMA will go offline and engine may flameout.
START (white)	Start relay is in START mode.	If START switch has not been engaged and there is zero indication on AMPS gage; START relay has malfunctioned and helicopter is on battery power. START circuit breaker — Out. Land as soon as practical.
T/R CHIP	Ferrous particles in tail rotor gearbox oil.	Land as soon as possible.
XMSN CHIP	Ferrous particles in transmission oil.	Land as soon as possible.

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Section 4

PERFORMANCE

4-1. INTRODUCTION

Performance data presented herein are derived from engine manufacture's specification power for engine less installation losses. These data are applicable to basic helicopter without any optional equipment that would appreciably affect lift, drag, or power available.

4-2. POWER ASSURANCE CHECK

A Power assurance check chart (figure 4-1) is provided for Allison model 250-C47B engine. This chart indicates maximum allowable MGT for an engine meeting minimum Allison specification. Engine must develop required torque without exceeding chart MGT in order to meet performance data contained in this manual.

Figure 4-1 may be used to periodically monitor engine performance.

To perform power assurance check, turn off all sources of bleed air, including ENGINE ANTI-ICING. Establish level flight at an AIRSPEED of 85 to 105 KIAS or V_{NE}, whichever is lower. Check may also be conducted in a hover prior to takeoff, depending on ambient conditions and gross weight.

Record following information from cockpit instruments:

EXAMPLE:

H_P	6000 feet
OAT	10°C
MGT	Actual reading
TORQUE	70%

SOLUTION:

Enter Power assurance check chart at observed TORQUE (70%), proceed vertically down to intersect H_P (6000 feet), follow horizontally to intersect indicated OAT (10°C), then drop vertically to read maximum allowable MGT.

If actual MGT is less than or equal to chart MGT, engine performance equals or exceeds minimum specification and performance data contained in this manual can be achieved.

If actual MGT is greater than chart MGT, engine performance is less than minimum specification and all performance data contained in this manual cannot be achieved. Refer to appropriate maintenance manual to determine cause of low power (high MGT).

NOTE

Chart may also be used to determine minimum specification power for actual MGT. Using above example, enter chart at actual MGT (675°C, proceed up to OAT (10°C), across to H_P (6000 feet), and up to read minimum torque available (70%). If actual power is equal to or greater than chart torque, engine performance equals or exceeds minimum specification and performance data contained in this manual can be achieved. If actual torque indication is less than chart torque, engine performance is less than minimum specification and all performance in this manual cannot be achieved. Refer to appropriate maintenance manual to determine cause of low power.

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4-3. DENSITY ALTITUDE

A Density altitude and temperature conversion chart (Figure 4-2) is provided to aid in calculation of performance and limitations. H_D is an expression of density of air in terms of height above sea level; hence, the less dense the air, the higher the H_D. For standard conditions of temperature and pressure, H_D is same as H_P. As temperature increases above standard for an altitude, H_D will also increase to values higher than H_P. Figure 4-2 expresses H_D as a function of H_P and temperature.

Density altitude chart also includes the inverse of the square root of the density ratio ($1/\sqrt{\sigma}$), which is used to calculate true airspeed by the following relation:

$KTAS = KCAS \times 1/\sqrt{\sigma}$

EXAMPLE:

If ambient temperature is -15°C and H_P is 7000 feet, find H_D, $1/\sqrt{\sigma}$, and true airspeed for 100 KCAS.

SOLUTION:

Enter bottom of chart at -15°C.

Move vertically upward to 7000 feet H_P line.

From this point, move horizontally to left and read H_D of 5000 feet, move horizontally to right and read $1/\sqrt{\sigma}$ = 1.07728.

True airspeed = KCAS $\times 1/\sqrt{\sigma}$ = 100 × 1.07728 = 107.7 KTAS.

4-4. HEIGHT - VELOCITY ENVELOPE

The Height - Velocity envelope charts (Figures 4-3 and 4-4) define conditions from which a safe landing can be made on a smooth, level, firm surface; following an engine failure. The Height - Velocity diagram (Figure 4-4) is valid only when helicopter gross weight does not exceed limits of the Altitude vs Gross Weight for Height - Velocity diagram (Figure 4-3). Four envelopes (Gross Weight Regions) are specified. Each Gross Weight Region applies for all gross weights within its boundaries. No interpolation is allowed.

For a given ambient outside air temperature, pressure altitude, and gross weight, the appropriate limiting envelope (Region A, B, C, or D) can be determined. Using Figure 4-3 (Altitude VS Gross Weight), move upward vertically from entry OAT to pressure altitude. From that point, move right horizontally to determine the correct weight region. (Examples: 15°C at Sea Level at 5000 pounds GW = Region B, and 30°C at 2000 feet pressure altitude at 5000 pounds GW = Region D) Once the correct weight region has been determined (A, B, C, or D), the corresponding Height - Velocity diagram is selected from Figure 4-4.

4-5. HOVER CEILING

NOTE

Hover performance charts are based on 100% ROTOR RPM.

Satisfactory stability and control have been demonstrated in each area of the Hover ceiling charts with winds as depicted on Hover ceiling wind accountability chart (Figure 4-5, 4-5A).

Hover ceiling - in ground effect charts (Figure 4-6) and Hover ceiling - out of ground effect charts (Figure 4-7) present hover performance as allowable gross weight for conditions of H_P and OAT. These hovering weights are obtainable in zero wind conditions. Each chart is divided into two areas. Area A (non shaded area) and Area B (shaded area).

For the data presented below 14,000 ft H_D, Area A of the hover ceiling chart presents hover performance (relative to GW) for conditions where adequate control margins exist for all relative wind conditions up to 35 knots for lateral CG not exceeding ±2.5 inches (±63 mm); and up to 17 knots, for lateral CG not exceeding ±4.0 inches (±102 mm); for hover, takeoff and landing. Area B of the hover ceiling charts presents hover

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performance (relative to GW) for conditions where adequate control margins exist for all relative winds within ±45° of the nose of the helicopter up to 35 knots for lateral CG not exceeding ±2.5 inches (±63 mm); for hover, take off, and landing.

For data presented above 14,000 ft H_D, Area A of the hover ceiling chart presents hover performance (relative to GW) for conditions where adequate control margins exist for all relative wind conditions up to 20 knots for lateral CG not exceeding ±2.5 inches (±63 mm); for hover, take off, and landing. Area B of the hover ceiling charts presents hover performance (relative to GW) where adequate control margins exist for relative winds within ±30° of the nose of the helicopter for hover, take off and landing, for lateral CG not exceeding ±2.5 inches (±63 mm).

EXAMPLE:

What IGE GW hover capability could be expected for the following conditions:

A. HEATER and ANTI ICE – OFF
B. HP – 6000 feet
C. OAT – +20 °C
D. TAKE OFF POWER

SOLUTION:

A. Use Hover ceiling IGE – take off power chart (Figure 4-6).
B. Enter OAT scale at +20 °C.
C. Move upward to 6000 feet Hp curve.
D. Move horizontally to +20 °C curve.
E. Drop down to read maximum external gross weight or 5400 pounds (IGE hover capability exceeds maximum internal GW of 5000 pounds).

4-6. NOT USED

4-7. CLIMB AND DESCENT

4-7-A. CLIMB

Rate of climb charts are presented for various combinations of power settings and ENGINE ANTI-ICING switch positions. Refer to Figures 4-8 and 4-9.

Recommended best rate of climb airspeed is 60 KIAS.

Reduce rate of climb data 100 feet per minute when operating with any combination of door(s) removed.

The following example is for use with Rate of climb chart at takeoff power. The example is typical for use with all other Rate of climb charts.

EXAMPLE:

Find the maximum rate of climb that can be attained using takeoff power under the following conditions:

ENGINE ANTI-ICING	OFF
OAT	10°C
HP	14,000 feet
GW	3500 pounds

SOLUTION:

Enter appropriate gross weight chart (3500 pounds, 1600 kilograms). At HP scale of 14000 feet proceed horizontally to temperature of 10°C. Drop down vertically and read a rate of climb of 1700 feet per minute.

4-7-B. AUTOROTATION

Refer to figure 4-10 for autorotational glide distance as a function of altitude.

4-8. AIRSPEED CALIBRATION

Refer to figure 4-11 for airspeed installation correction during level flight and climb

4-9. NOT USED

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4-10. NOISE LEVELS

4-10-A. FAR PART 36 STAGE 2 NOISE LEVEL

This aircraft is certified as a Stage 2 helicopter as prescribed in FAR Part 36, Subpart H, for gross weights up to and including the certificated maximum takeoff and landing weight of 5000 pounds (2268 kilograms). There are no operating limitations to meet any of the noise requirements.

The following noise level complies with FAR Part 36, Appendix J, Stage 2 noise level requirements. It was obtained by analysis of approved data from noise tests conducted under the provisions of FAR Part 36, Amendment 36-20.

The certified flyover noise level for the Model 407 is 85.1 dBA SEL.

NOTE

No determination has been made by the certifying authorities that the noise levels of this aircraft are or should be acceptable or unacceptable for operations at, into, or out of any airport.

VH is defined as the airspeed in level flight obtained using the minimum specification engine torque corresponding to maximum continuous power available for sea level, 25°C (77°F) ambient conditions at the relevant maximum certificated weight. The value of VH thus defined for this aircraft is 127 KTAS.

4-10-B. CANADIAN AIRWORTHINESS MANUAL CHAPTER 516 AND ICAO ANNEX 16 NOISE LEVEL

This aircraft complies with the noise emission standards applicable to the aircraft as set out by the International Civil Aviation Organization (ICAO) in Annex 16, Volume 1, Chapter 11, for gross weights up to and including the certificated maximum takeoff and landing weight of 5000 pounds (2268 kilograms). There are no operating limitations to meet any of the noise requirements.

The following noise level complies with ICAO Annex 16, Volume 1, Chapter 11 noise level requirements. It was obtained by analysis of approved data from noise tests conducted under the provisions of ICAO Annex 16, Volume 1, Third Edition-1993.

The flyover noise level for the Model 407 is 84.6 dBA SEL.

NOTE

ICAO Annex 16, Volume 1, Chapter 11 approval is applicable only after endorsement by the Civil Aviation Authority of the country of aircraft registration.

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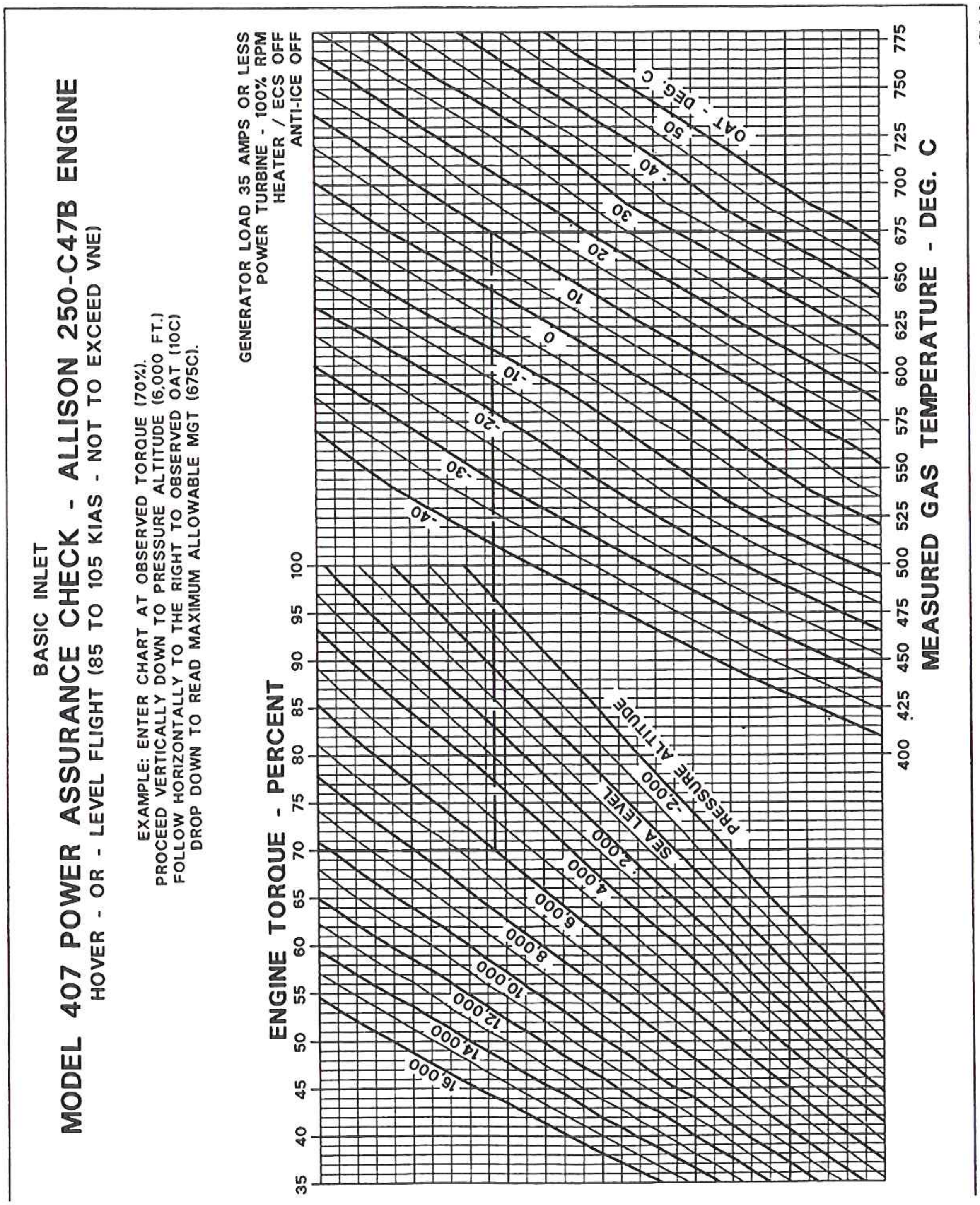

Figure 4-1. Power assurance check

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DENSITY ALTITUDE

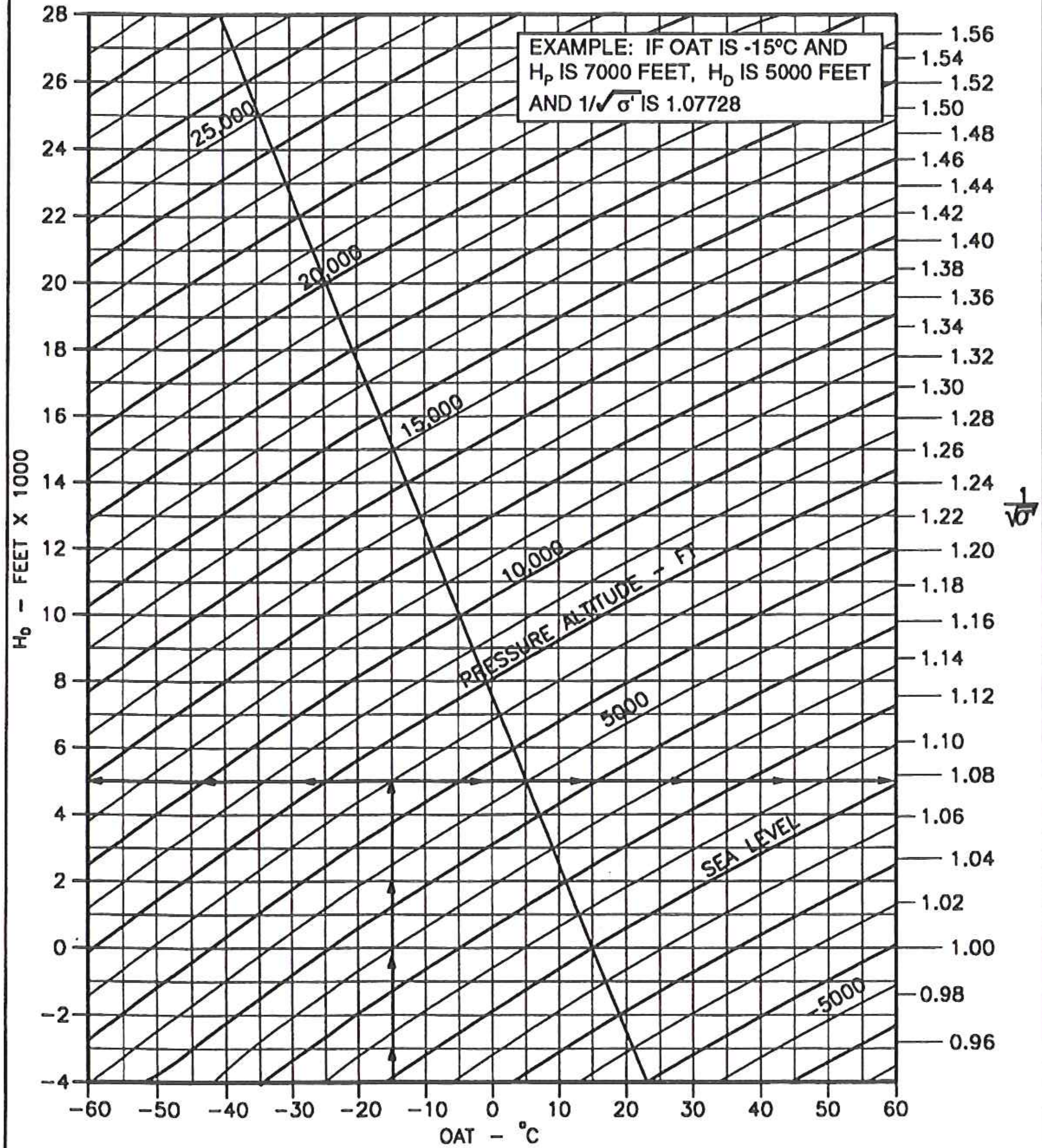

Figure 4-2. Density altitude

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M407
ALTITUDE VS GROSS WEIGHT
FOR HEIGHT - VELOCITY DIAGRAM

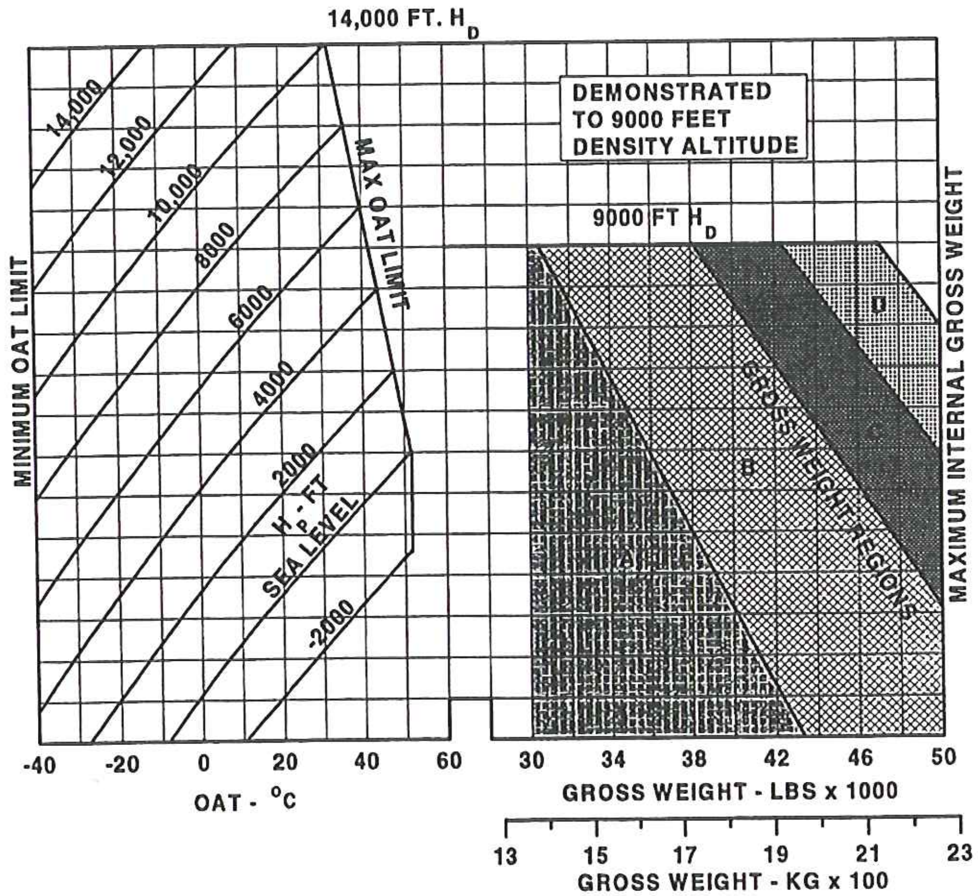

Figure 4-3. Altitude vs gross weight for height - velocity diagram

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HEIGHT - VELOCITY DIAGRAM
FOR GROSS WEIGHT REGIONS A TO D

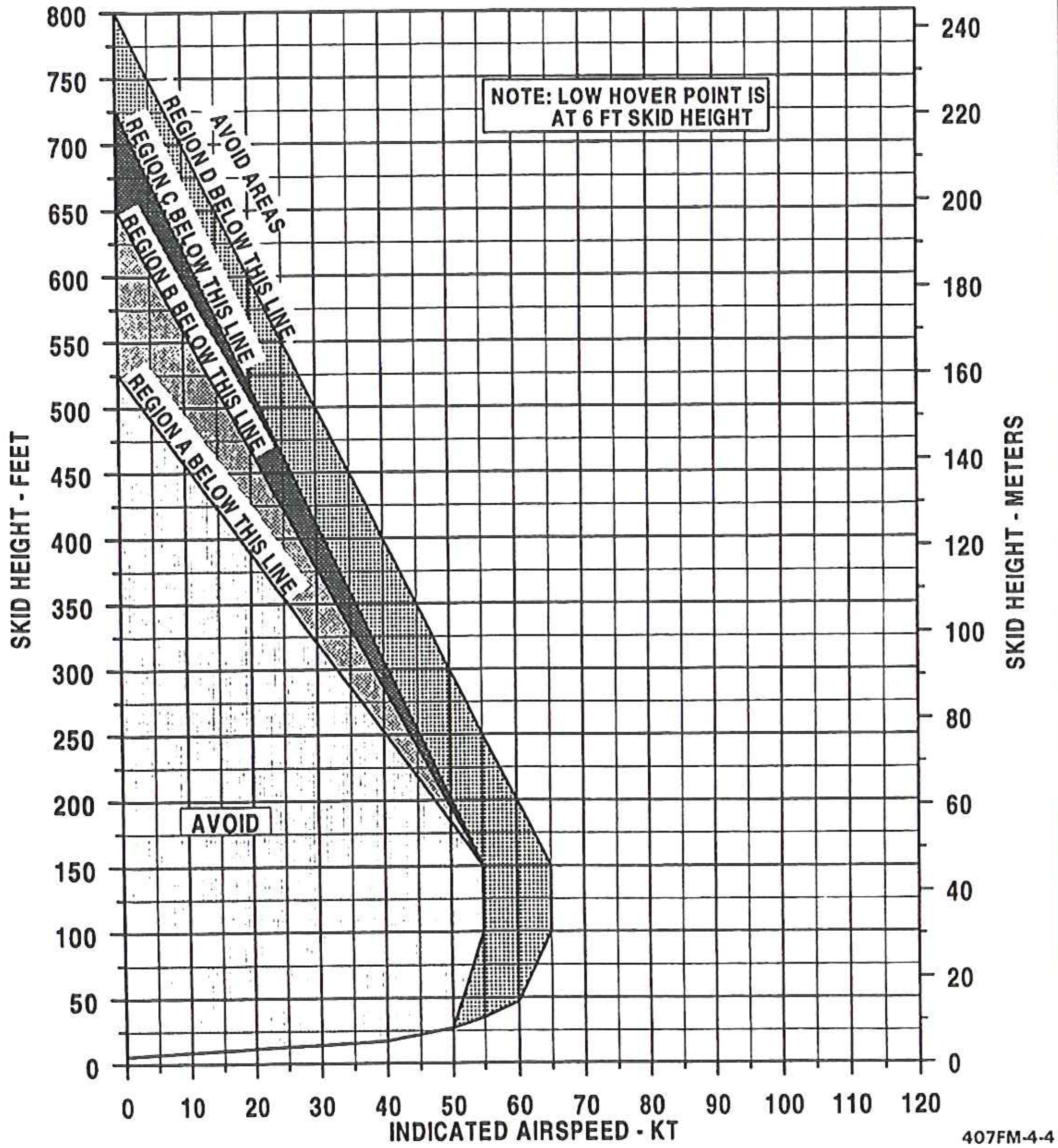

Figure 4-4. Height - velocity diagram

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HEIGHT - VELOCITY DIAGRAM FOR GROSS WEIGHT REGION B

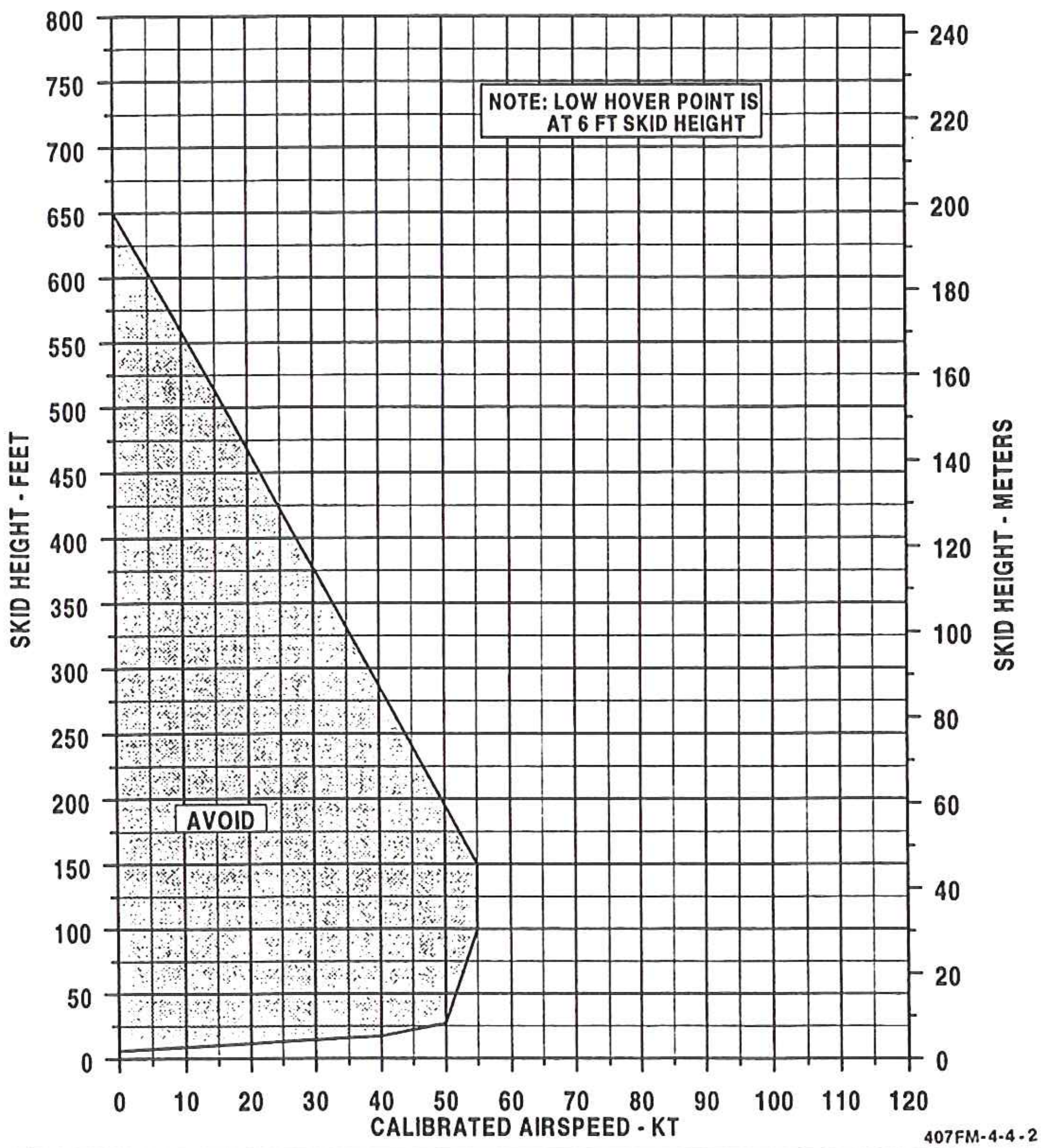

Figure 4-4. Height - velocity diagram (sheet 2 of 4)

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HEIGHT - VELOCITY DIAGRAM FOR GROSS WEIGHT REGION C

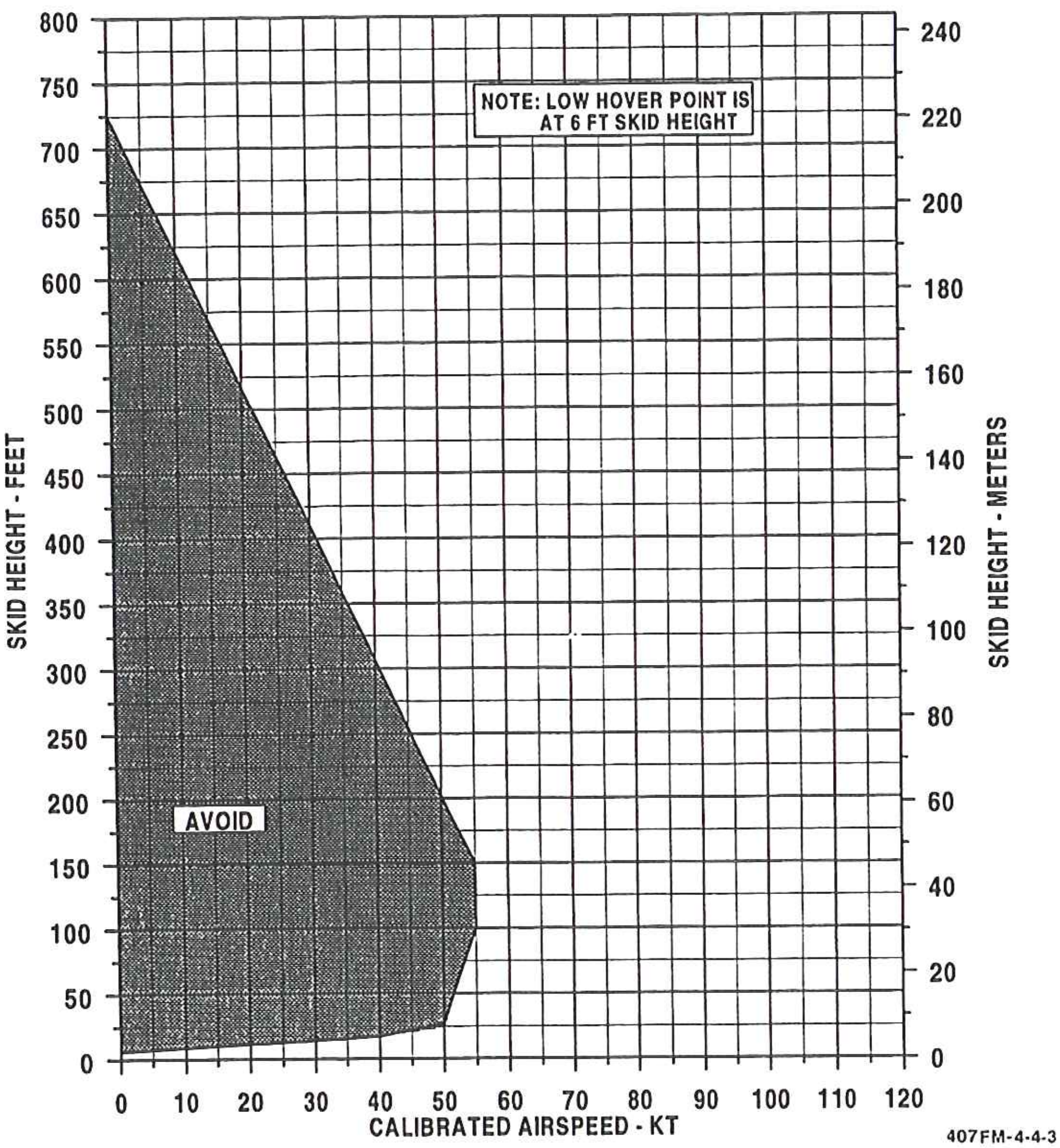

Figure 4-4. Height - velocity diagram (sheet 3 of 4)

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HEIGHT - VELOCITY DIAGRAM
FOR GROSS WEIGHT REGION D

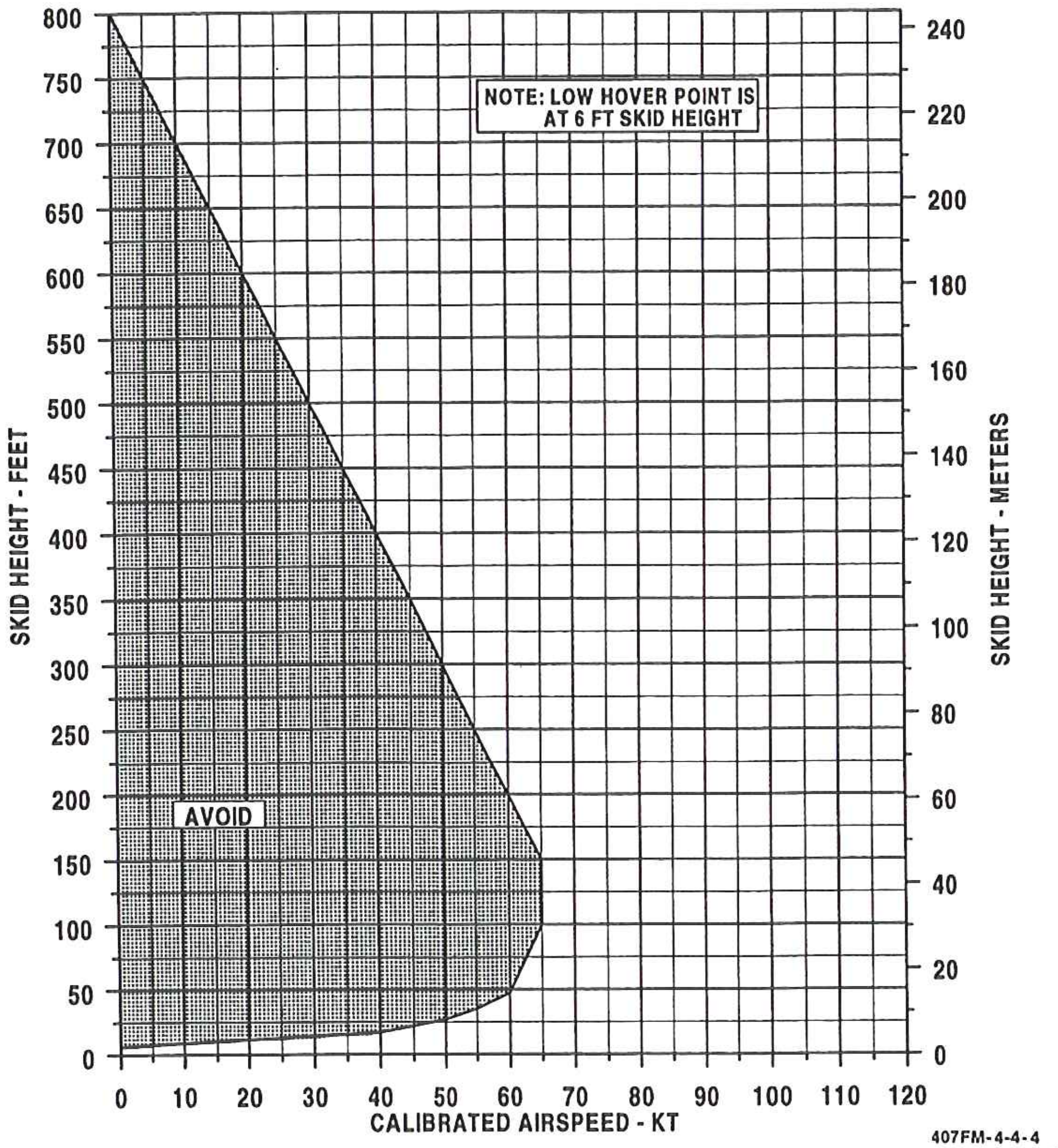

Figure 4-4. Height - velocity diagram (sheet 4 of 4)

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ALTITUDE UP TO 14000 FT H_D

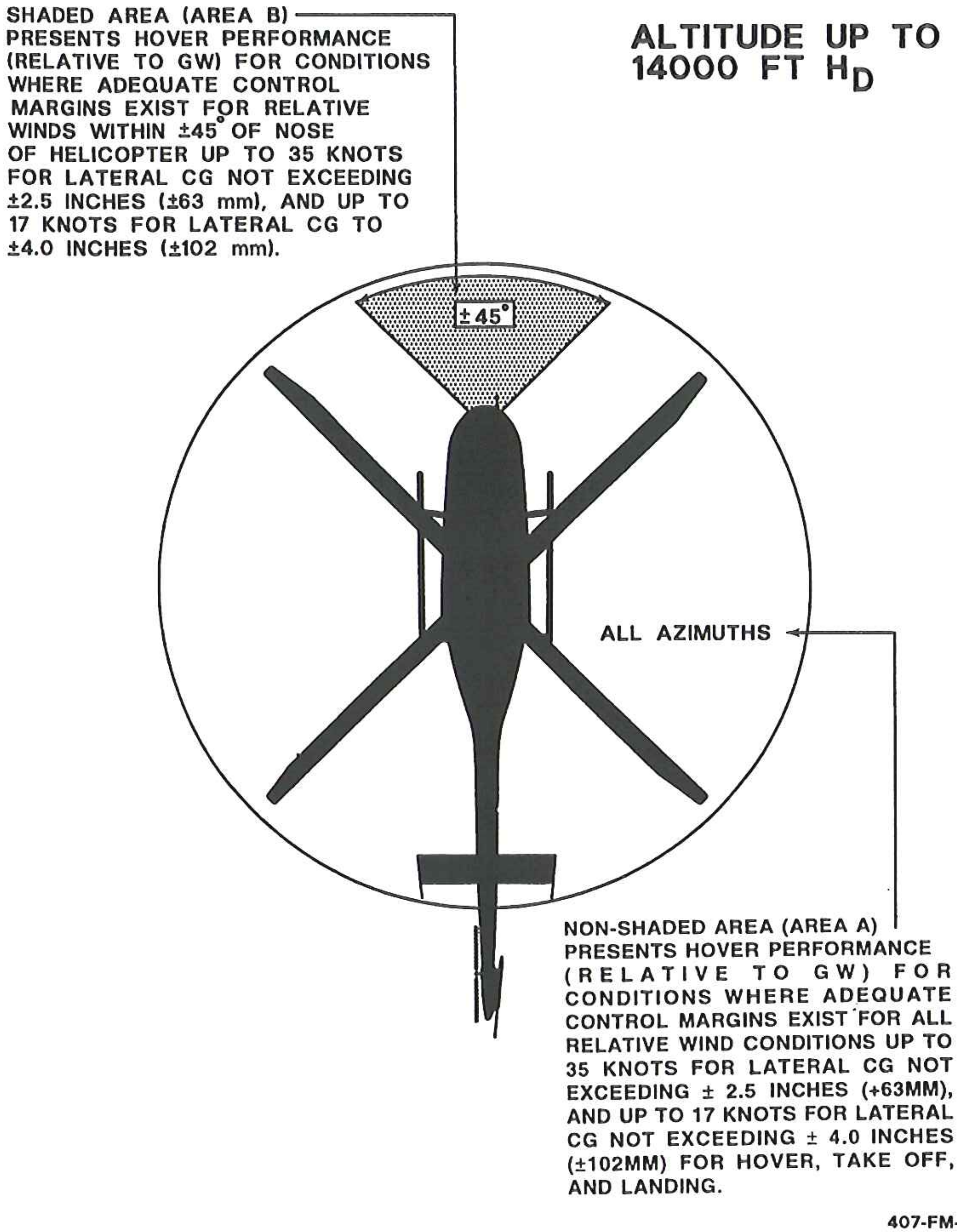

Figure 4-5. Hover ceiling wind accountability chart — below 14,000 feet H_D.

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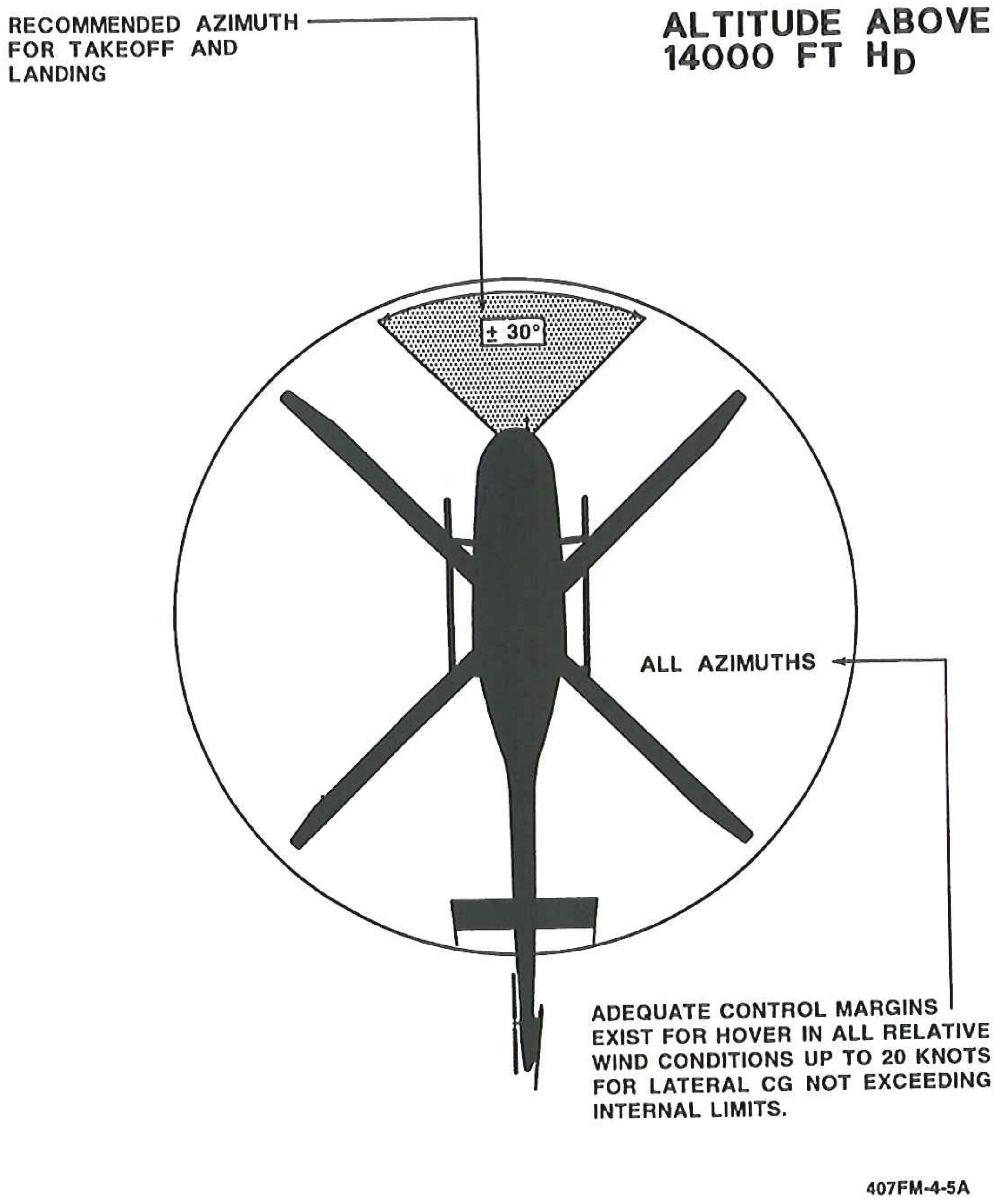

Figure 4-5A. Hover ceiling wind accountability chart — above 14,000 feet H_D.

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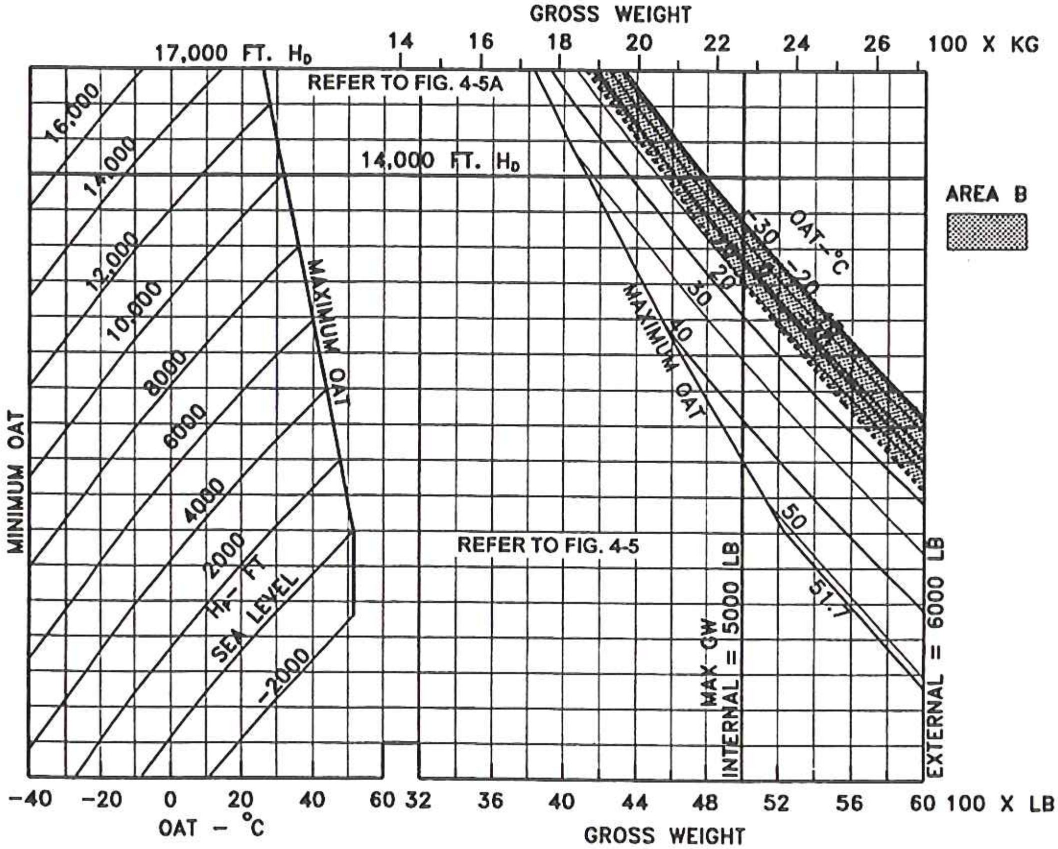
HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET



407FM-4-6-1

Figure 4-6. Hover ceiling IGE (sheet 1 of 8)

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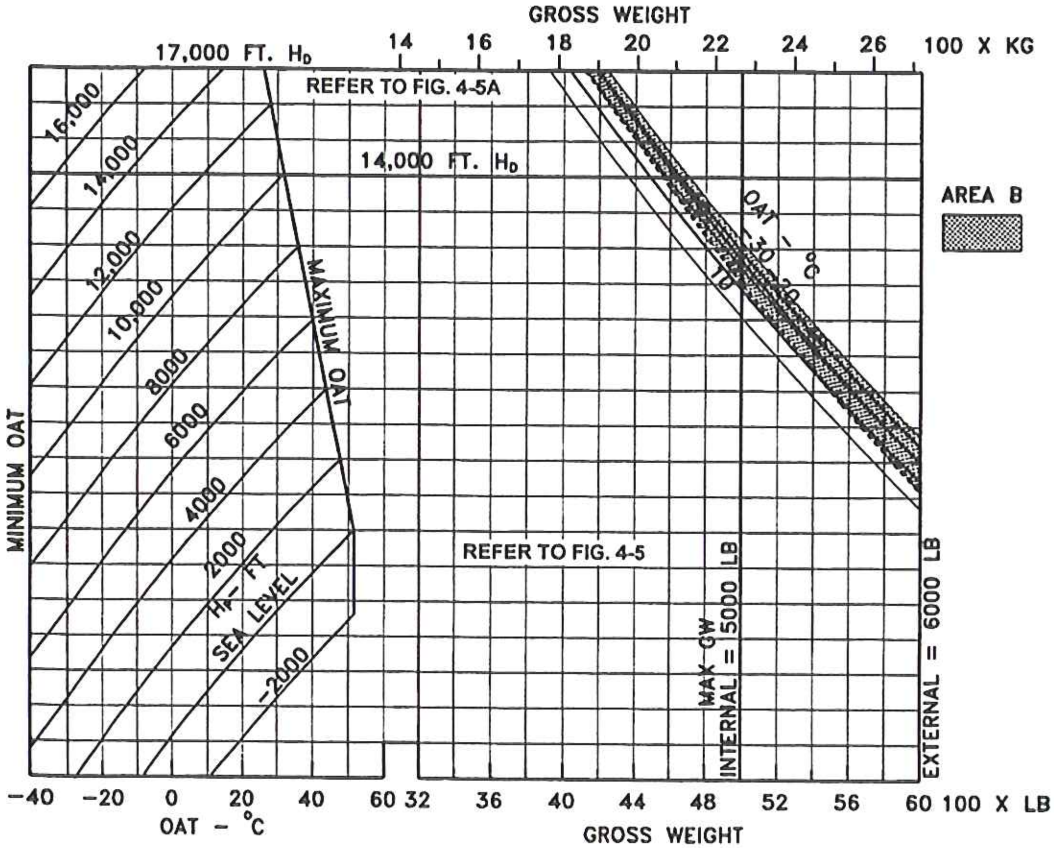
HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
BASIC INLET



407FM-4-6-2

Figure 4-6. Hover ceiling IGE (sheet 2 of 8)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
BASIC INLET

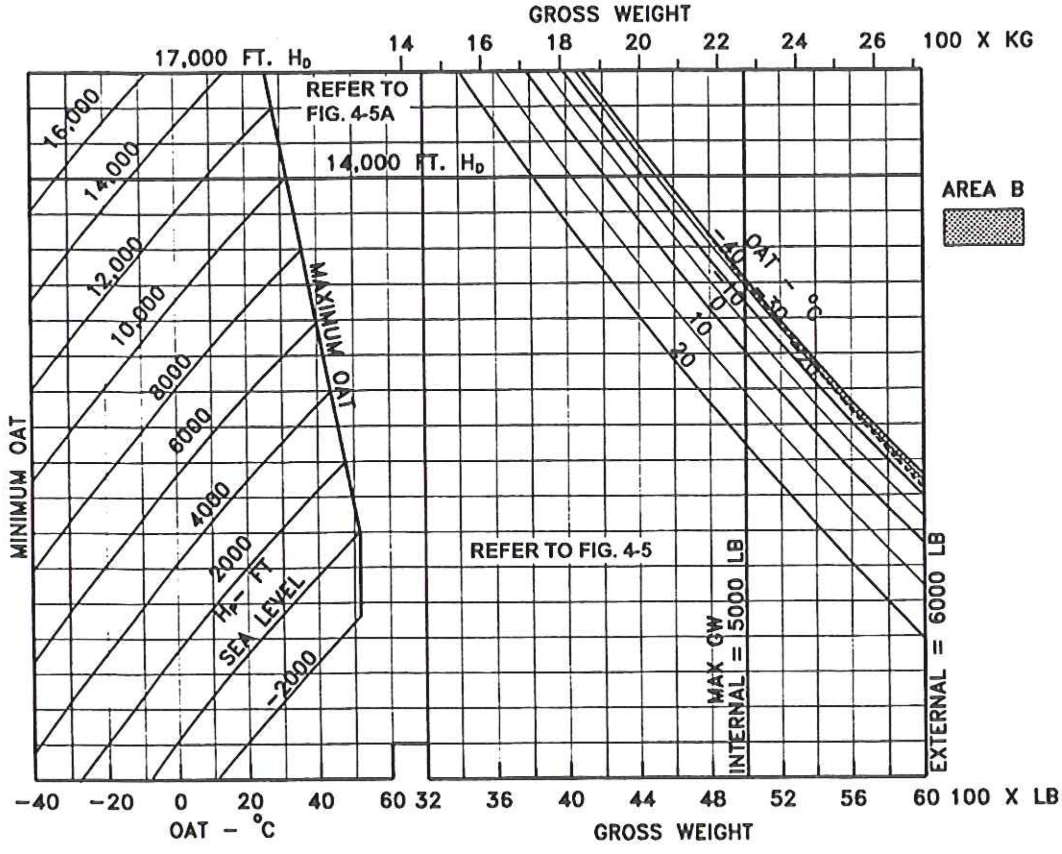

Figure 4-6. Hover ceiling IGE (sheet 3 of 8)

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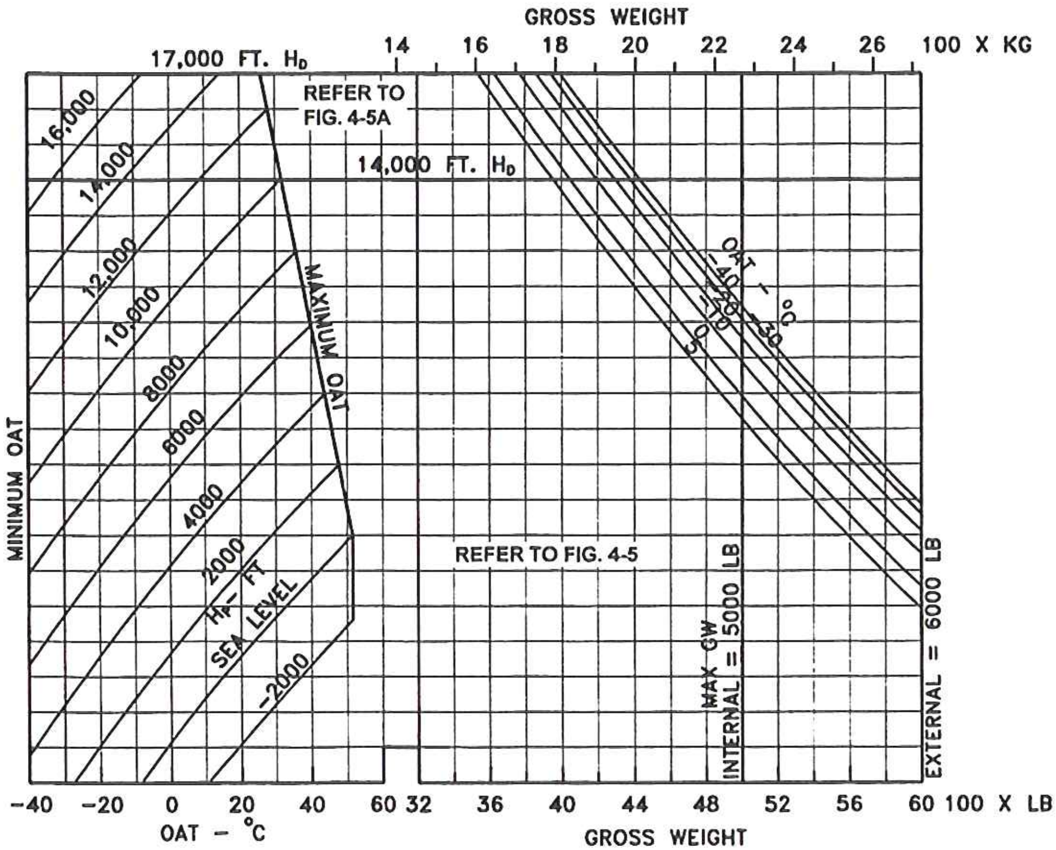
HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET



407FM-4-6-4

Figure 4-6. Hover ceiling IGE (sheet 4 of 8)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET

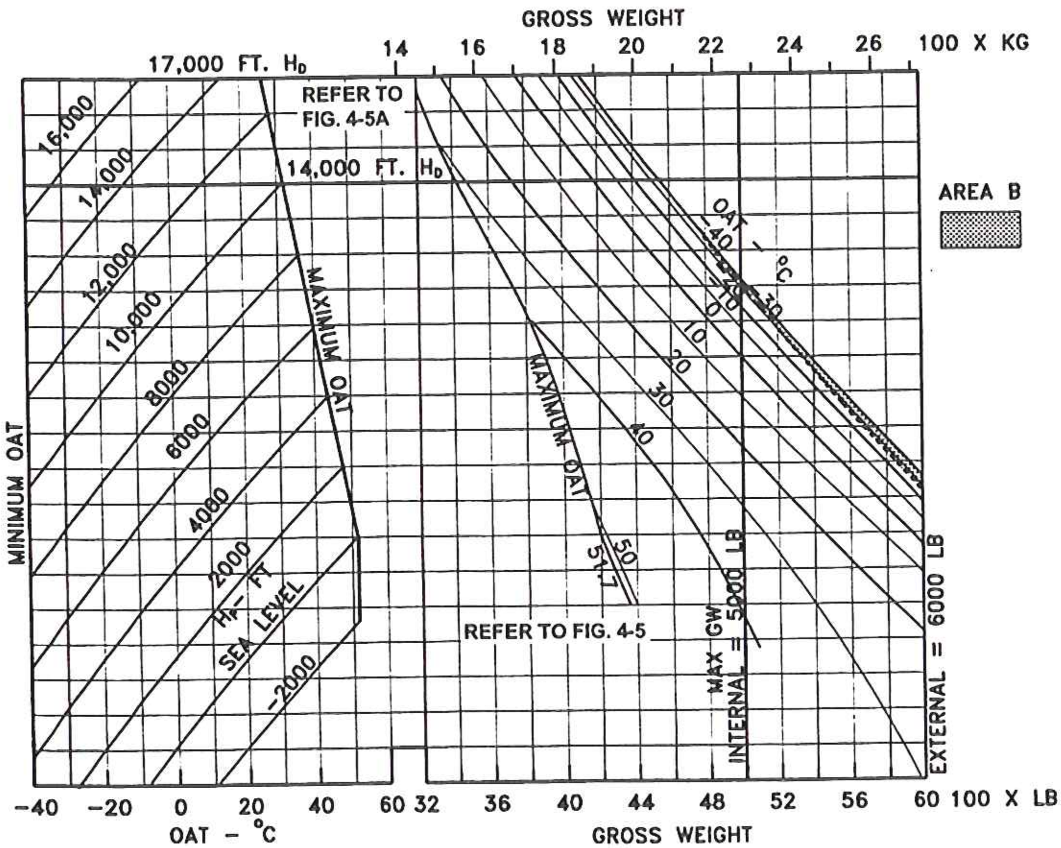

Figure 4-6. Hover ceiling IGE (sheet 5 of 8)

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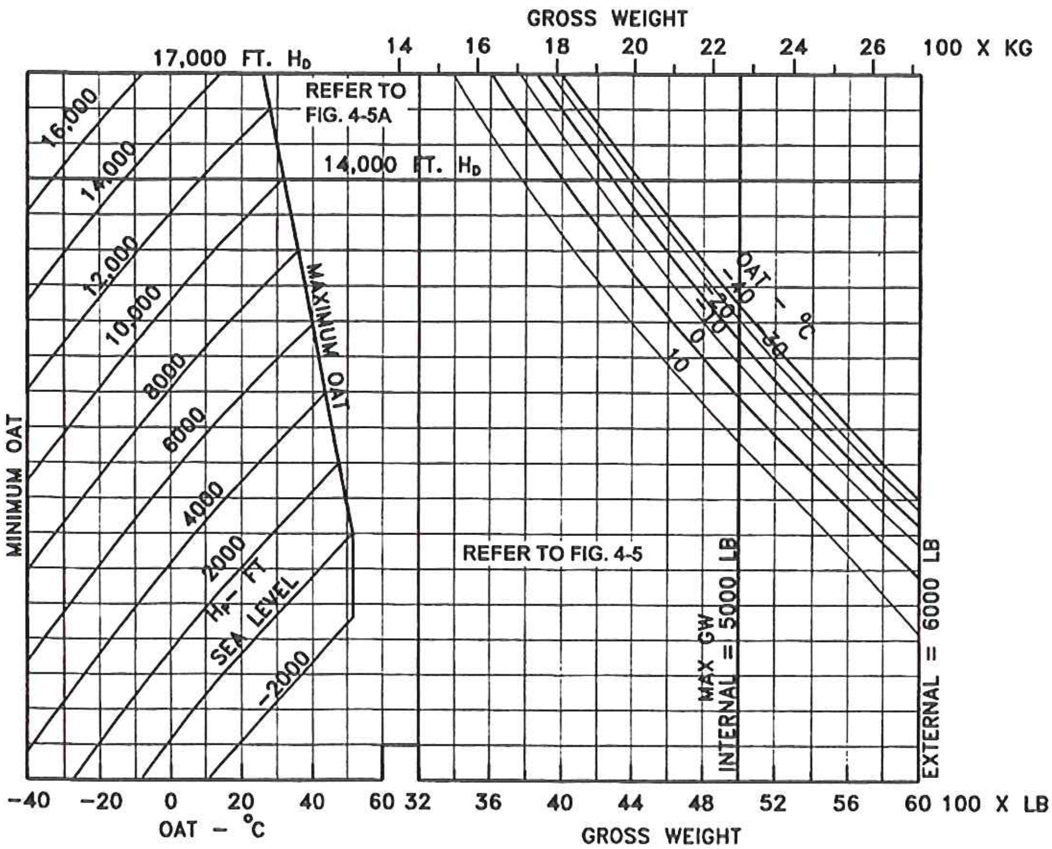
HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
BASIC INLET



407FM-4-6-6

Figure 4-6. Hover ceiling IGE (sheet 6 of 8)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
BASIC INLET

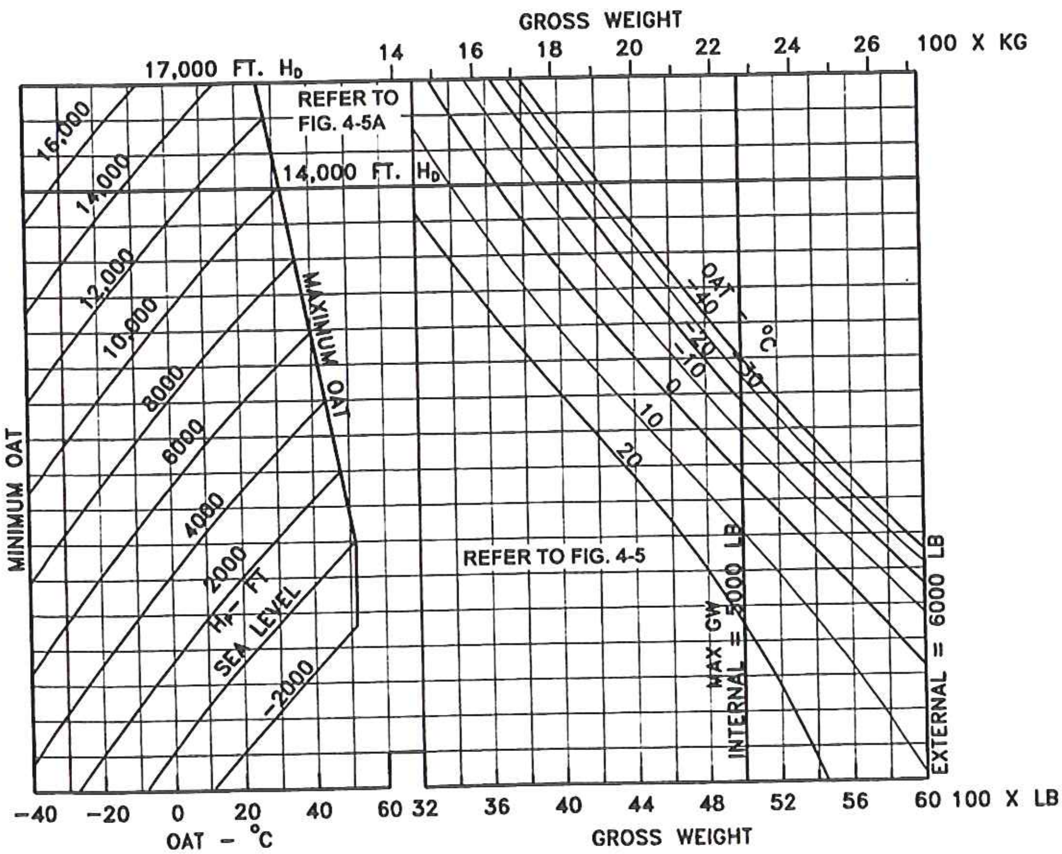

Figure 4-6. Hover ceiling IGE (sheet 7 of 8)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET

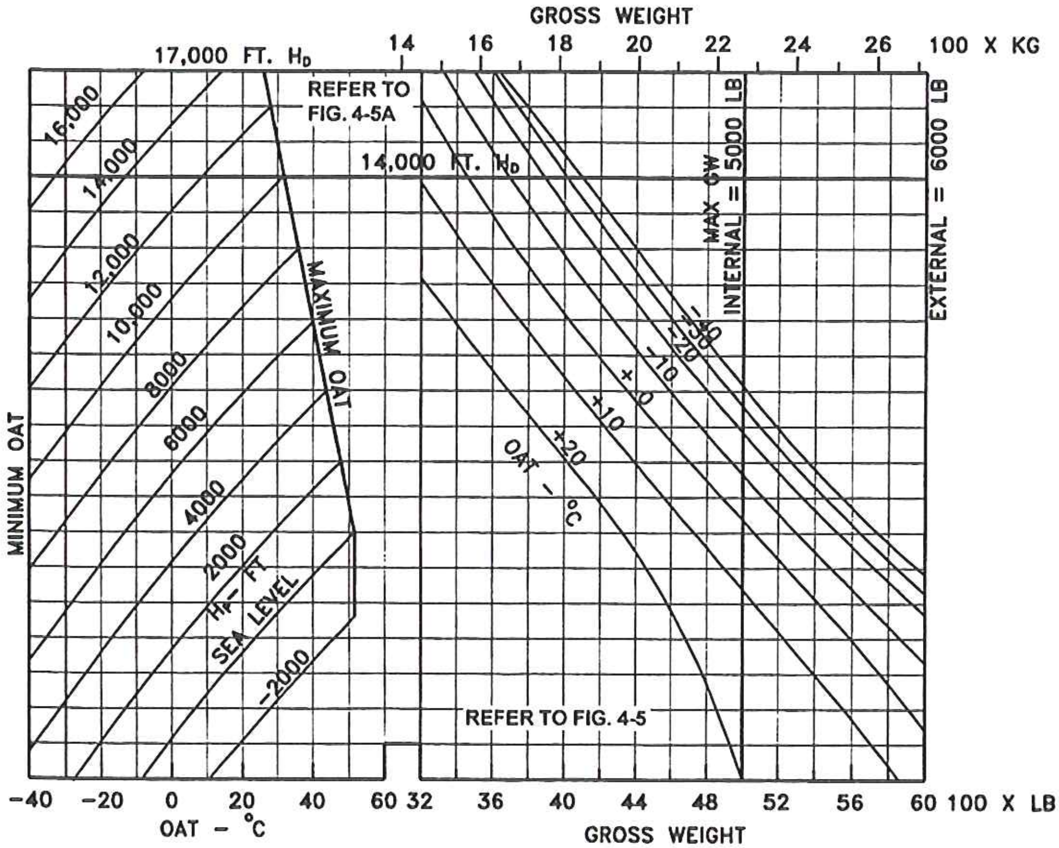

Figure 4-6. Hover ceiling IGE (sheet 8 of 8)

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HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET

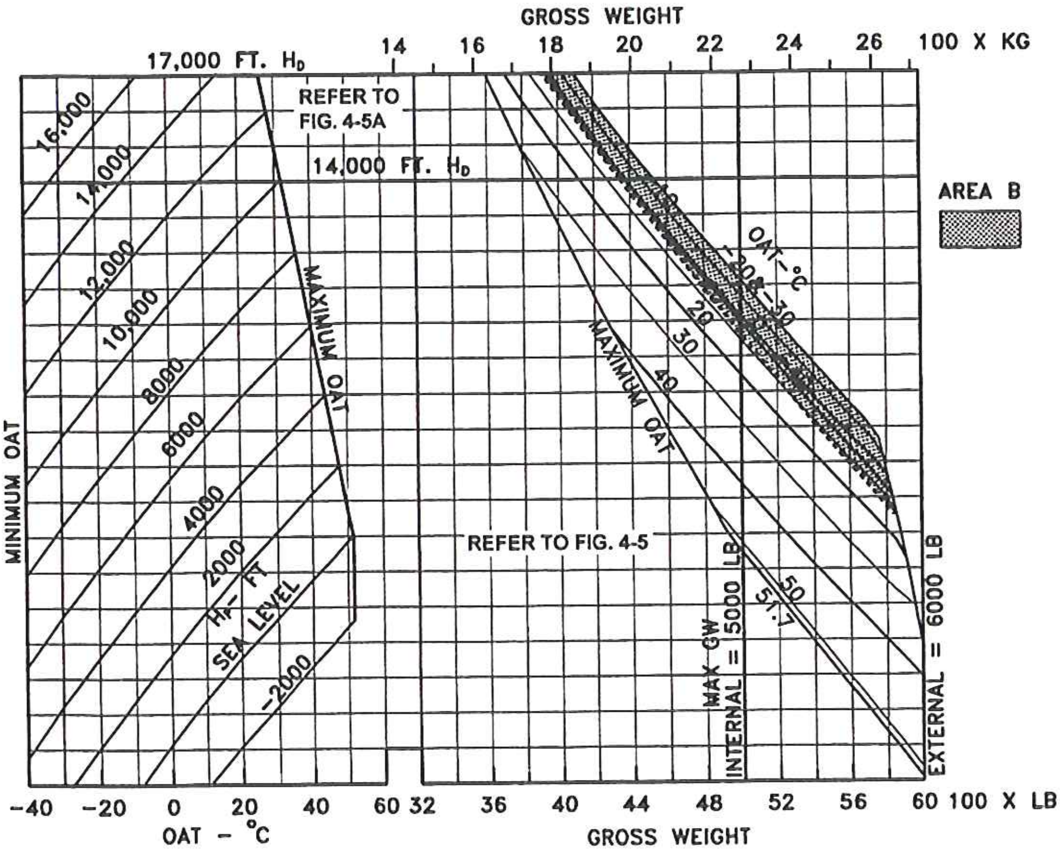

Figure 4-7. Hover ceiling OGE (sheet 1 of 8)

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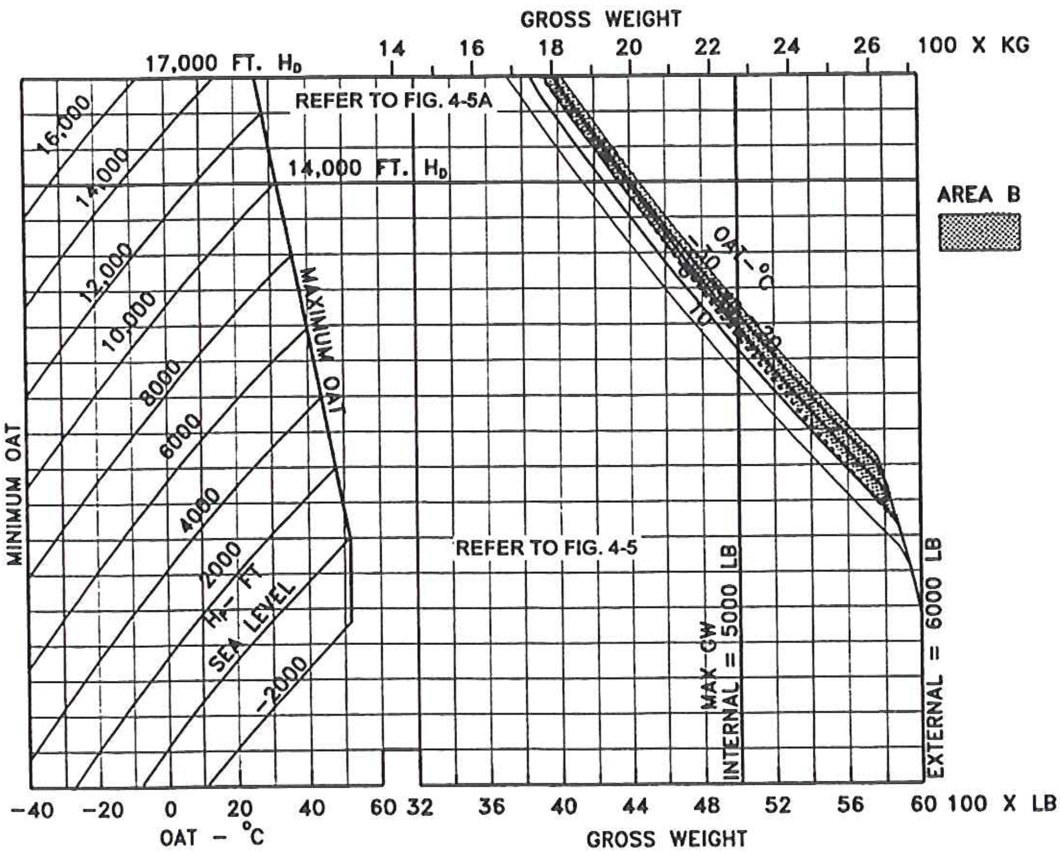
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
BASIC INLET



407FM-4-7-2

Figure 4-7. Hover ceiling OGE (sheet 2 of 8)

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HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
BASIC INLET

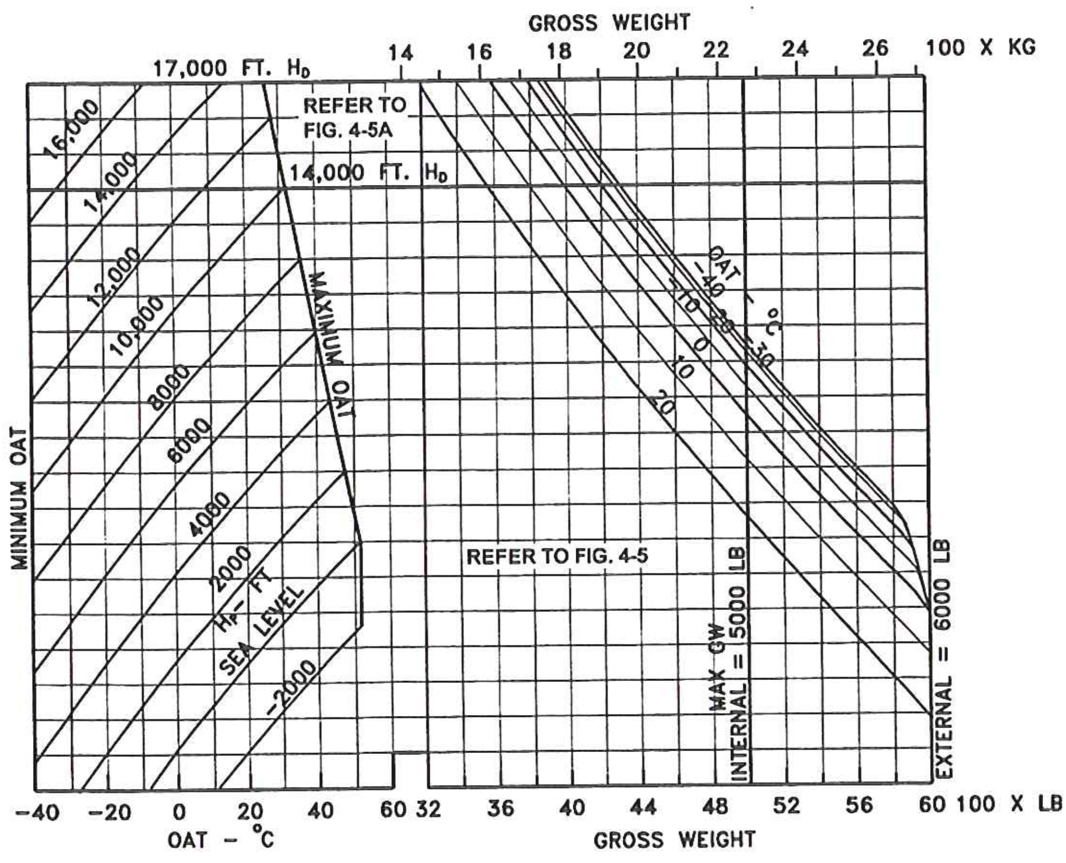

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Figure 4-7. Hover ceiling OGE (sheet 3 of 8)

407 PG Performance 08-31-2002
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HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET

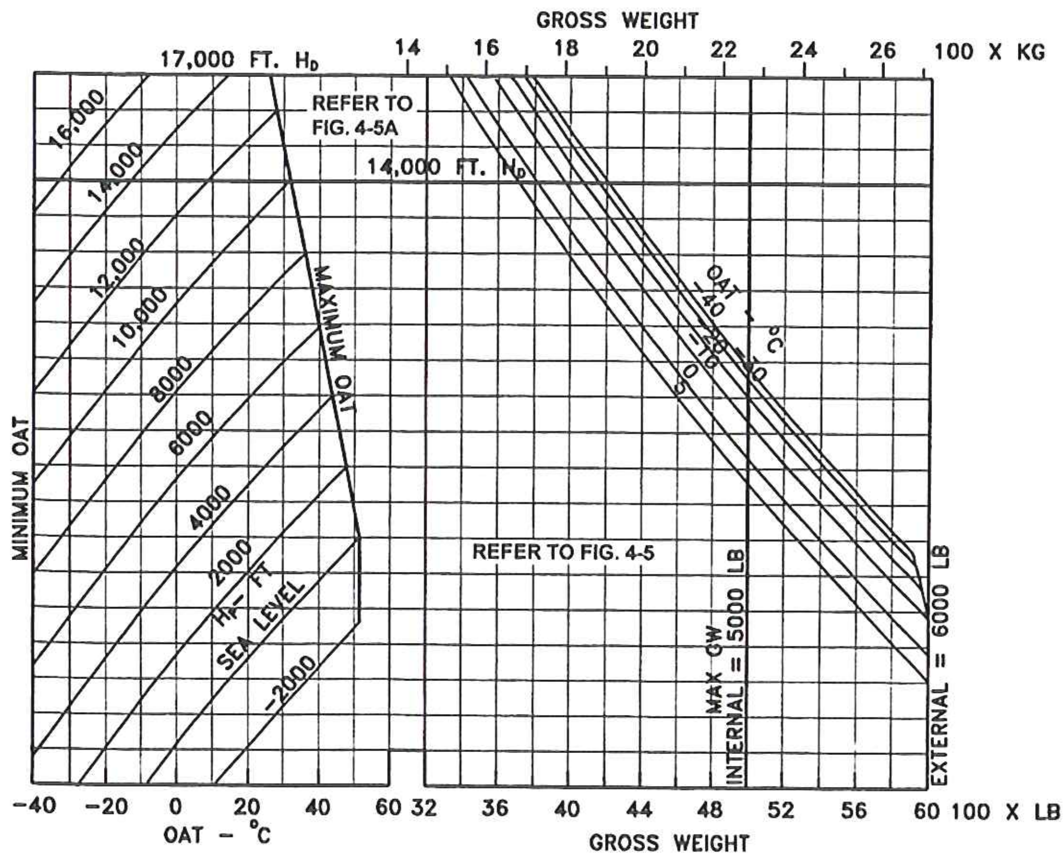

Figure 4-7. Hover ceiling OGE (sheet 4 of 8)

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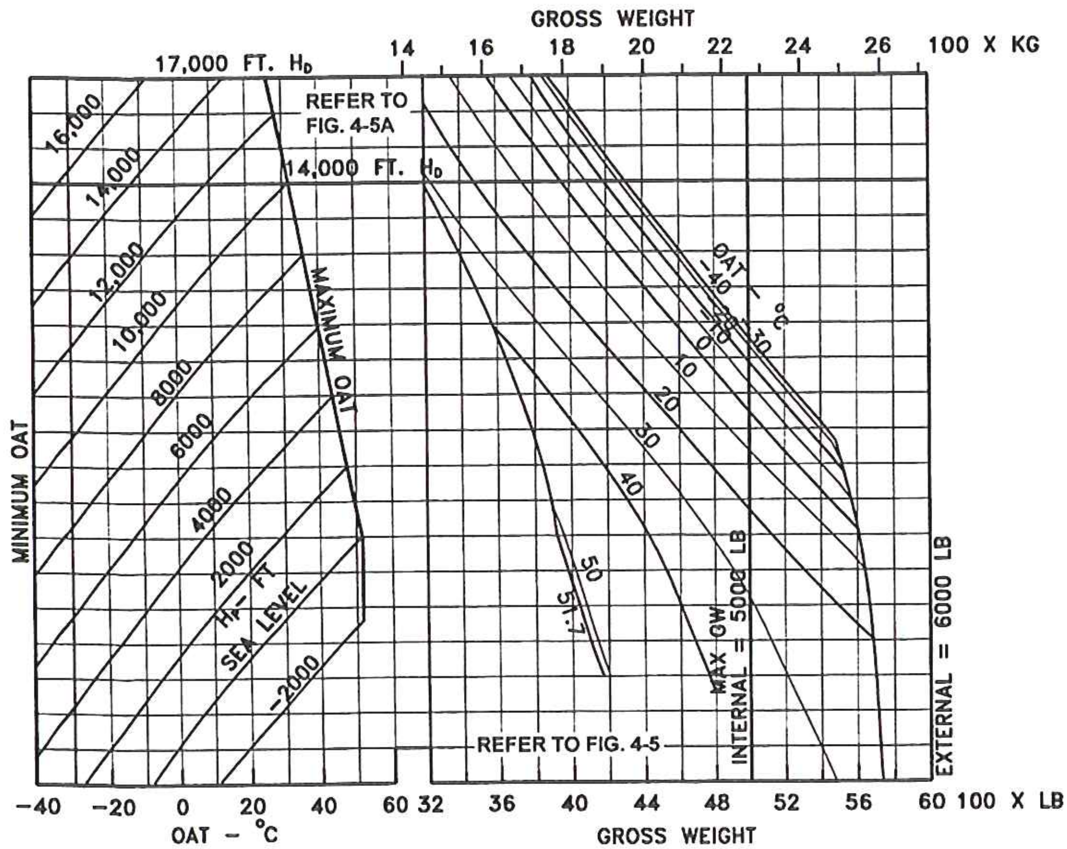
HOVER CEILING
OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET



407FM-4-7-5

Figure 4-7. Hover ceiling OGE (sheet 5 of 8)

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HOVER CEILING
OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
BASIC INLET

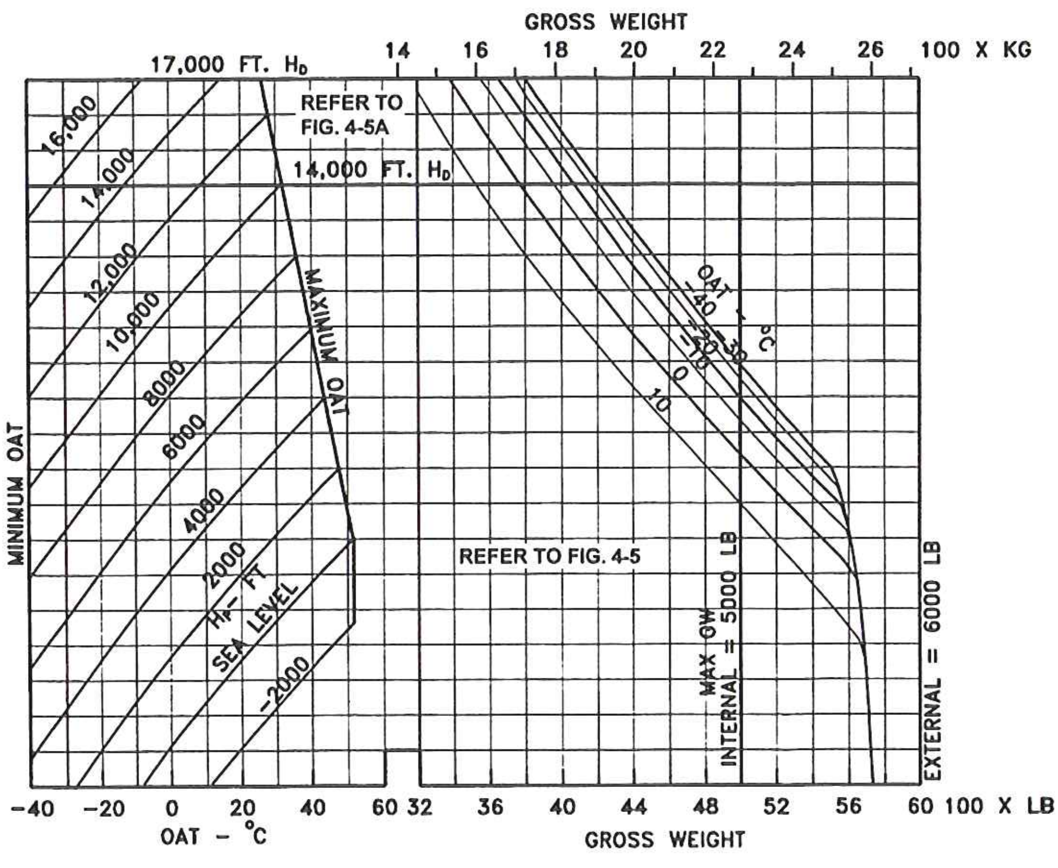

Figure 4-7. Hover ceiling OGE (sheet 6 of 8)

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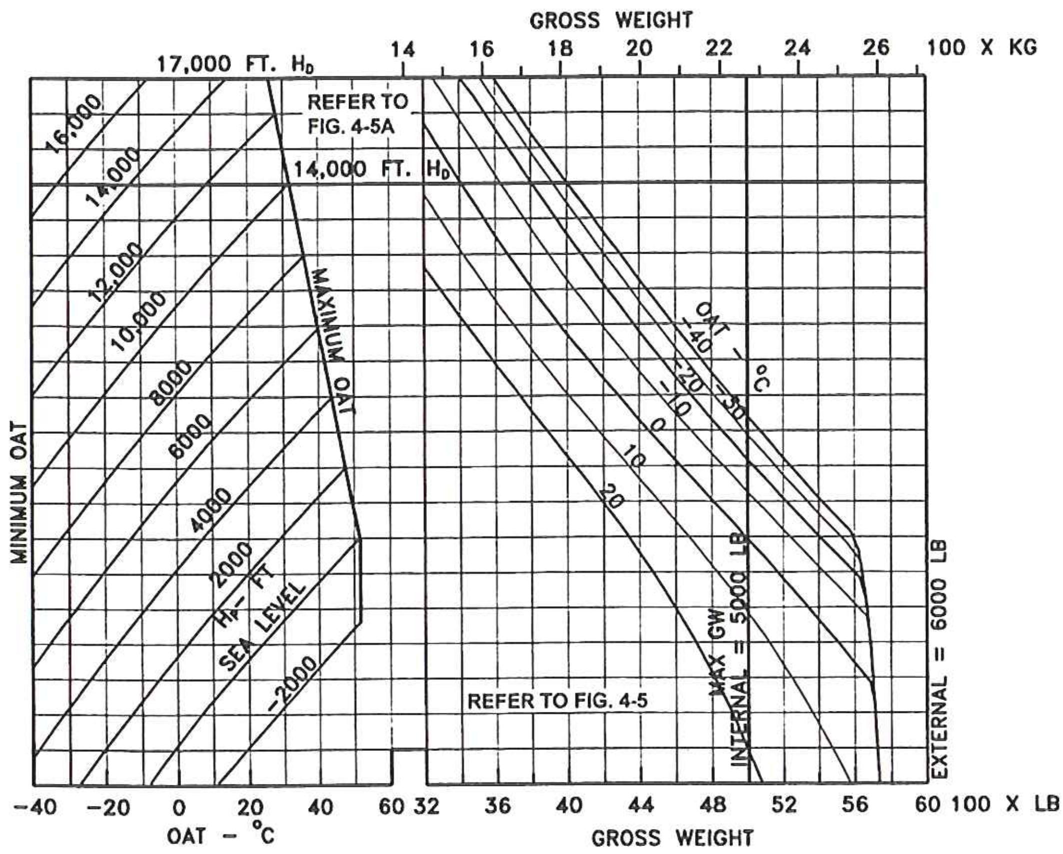
HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
BASIC INLET



407FM-4-7-7

Figure 4-7. Hover ceiling OGE (sheet 7 of 8)

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HOVER CEILING
OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET

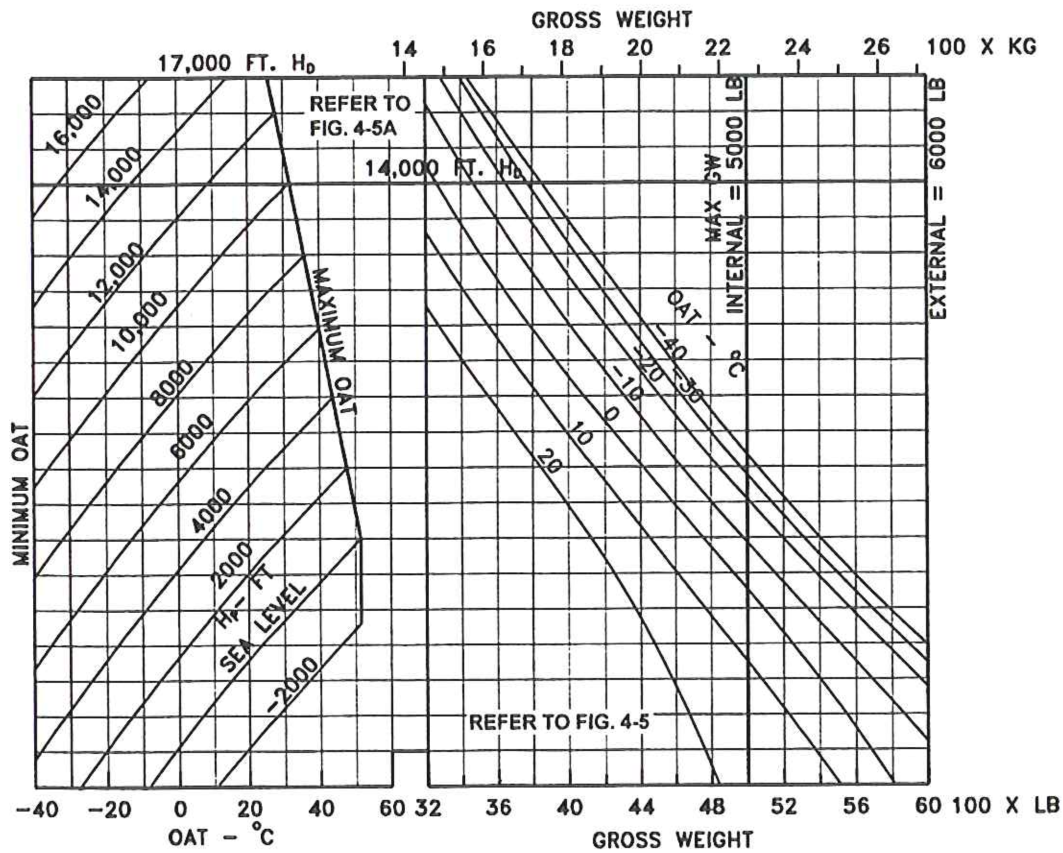

Figure 4-7. Hover ceiling OGE (sheet 8 of 8)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 180 FT/MIN ABOVE
15,000 FT H_P FOR ANTI-ICE ON (5°C AND COLDER)

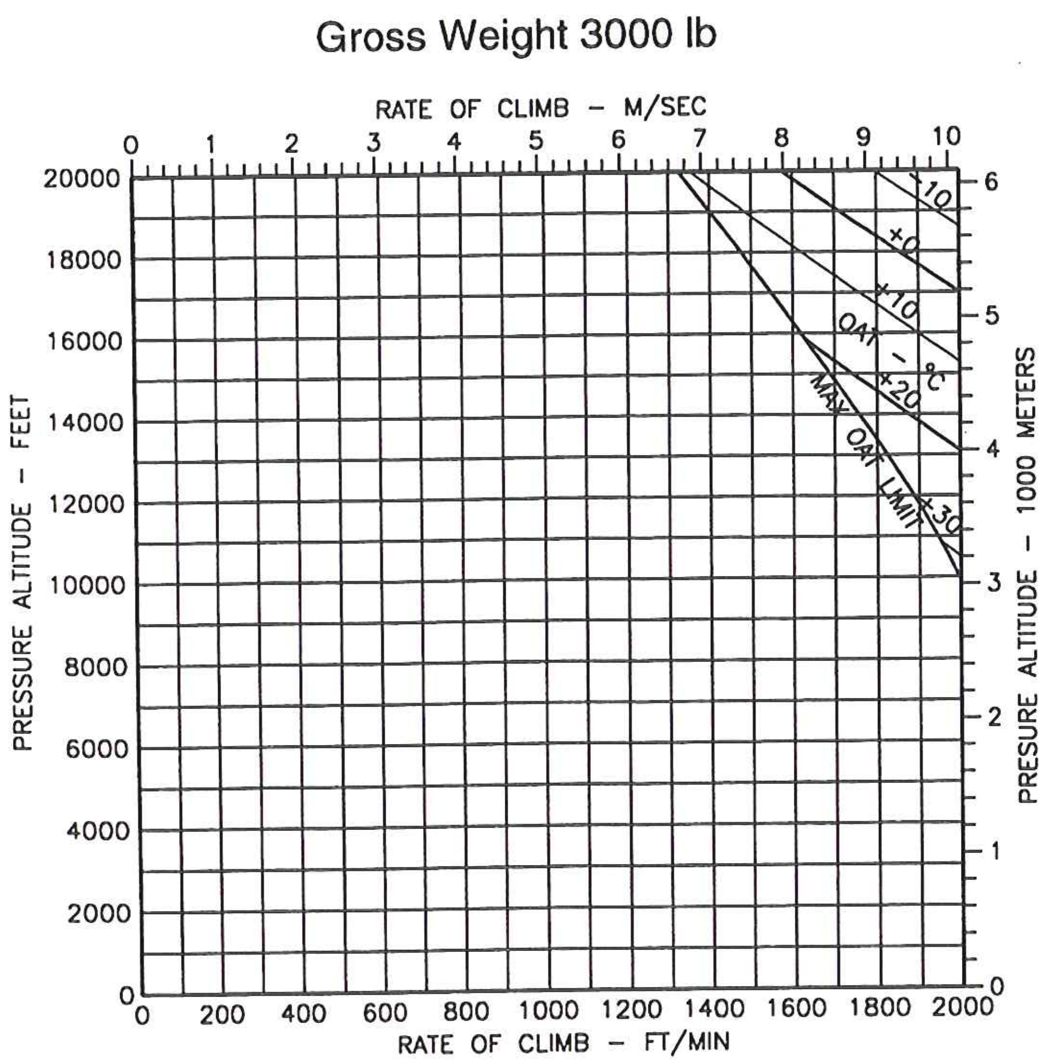

Figure 4-8. Rate of climb - takeoff power (sheet 1 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 155 FT/MIN ABOVE
11,500 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

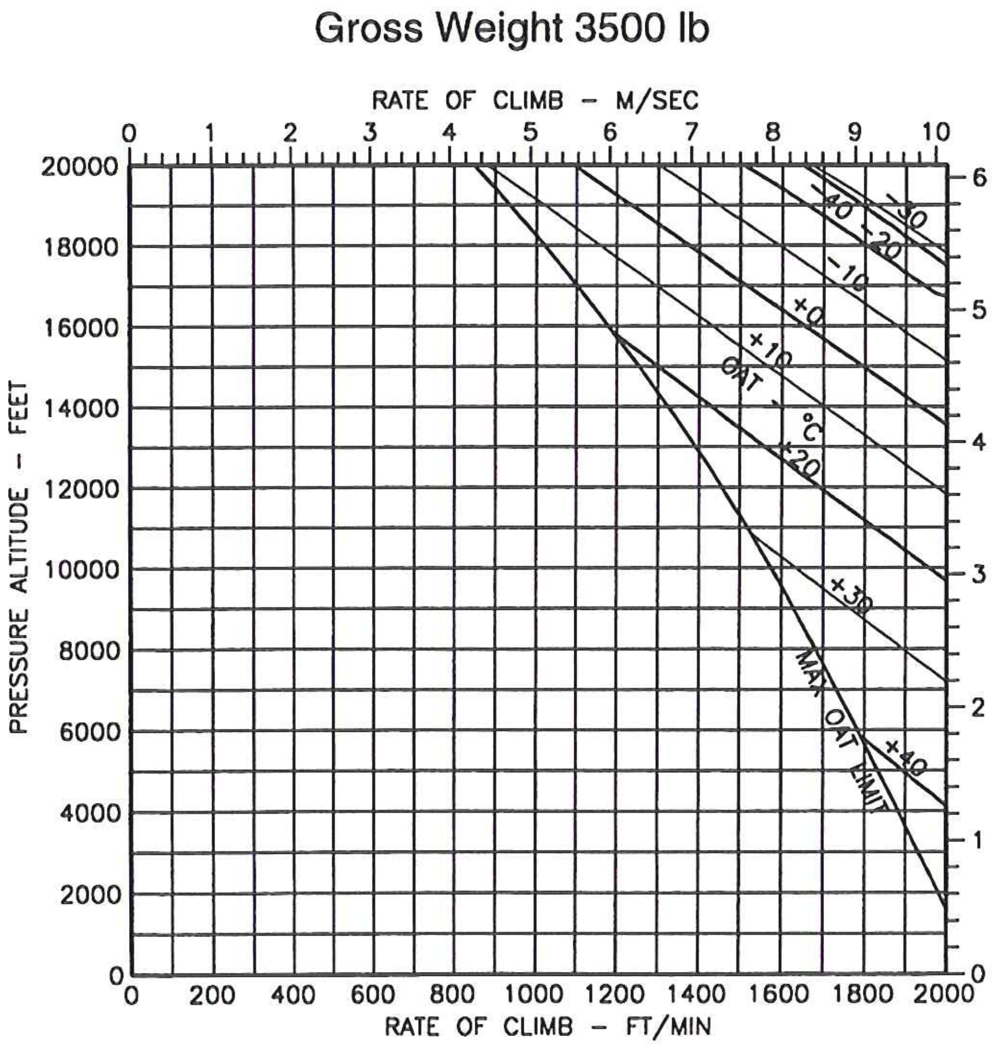

Figure 4-8. Rate of climb - takeoff power (sheet 2 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 135 FT/MIN ABOVE 9000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

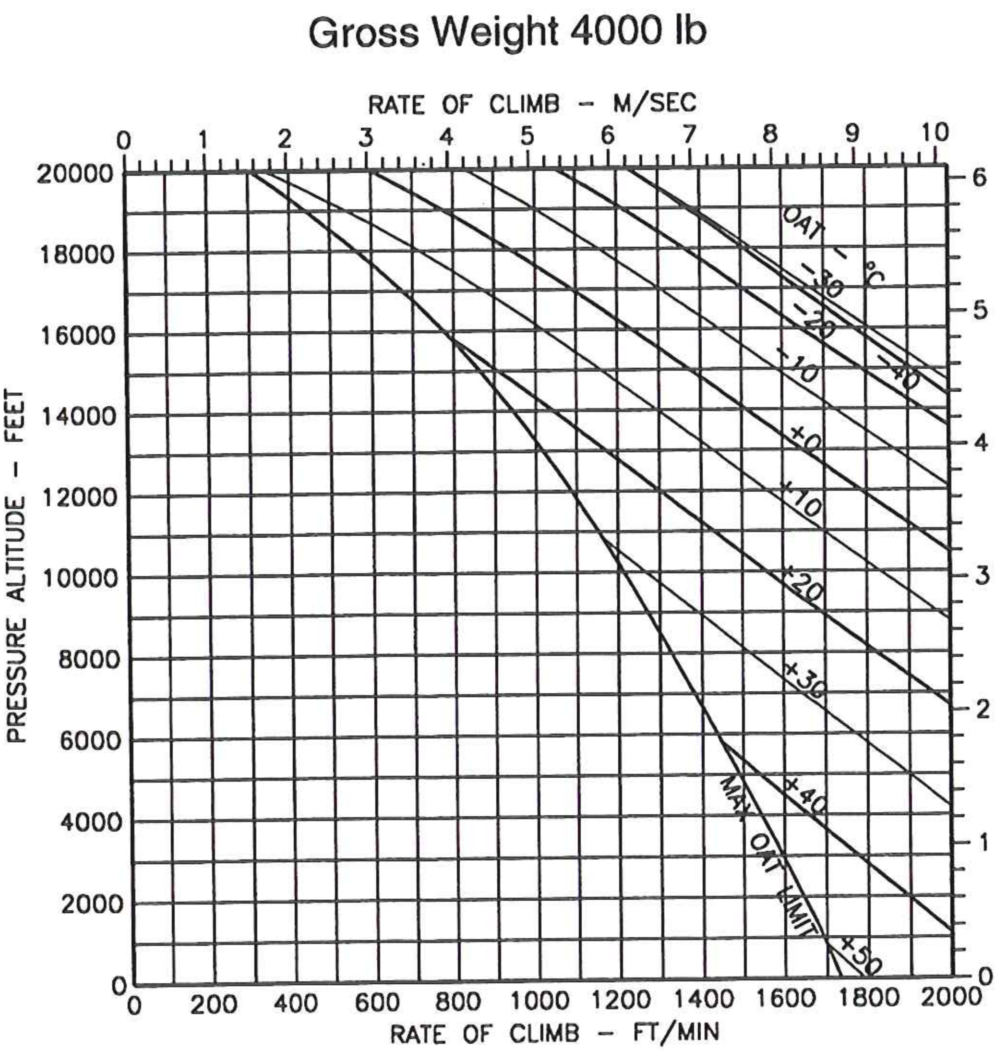

Figure 4-8. Rate of climb - takeoff power (sheet 3 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 120 FT/MIN ABOVE
6500 FT H_P FOR ANTI-ICE ON (5°C AND COLDER)

Gross Weight 4500 lb

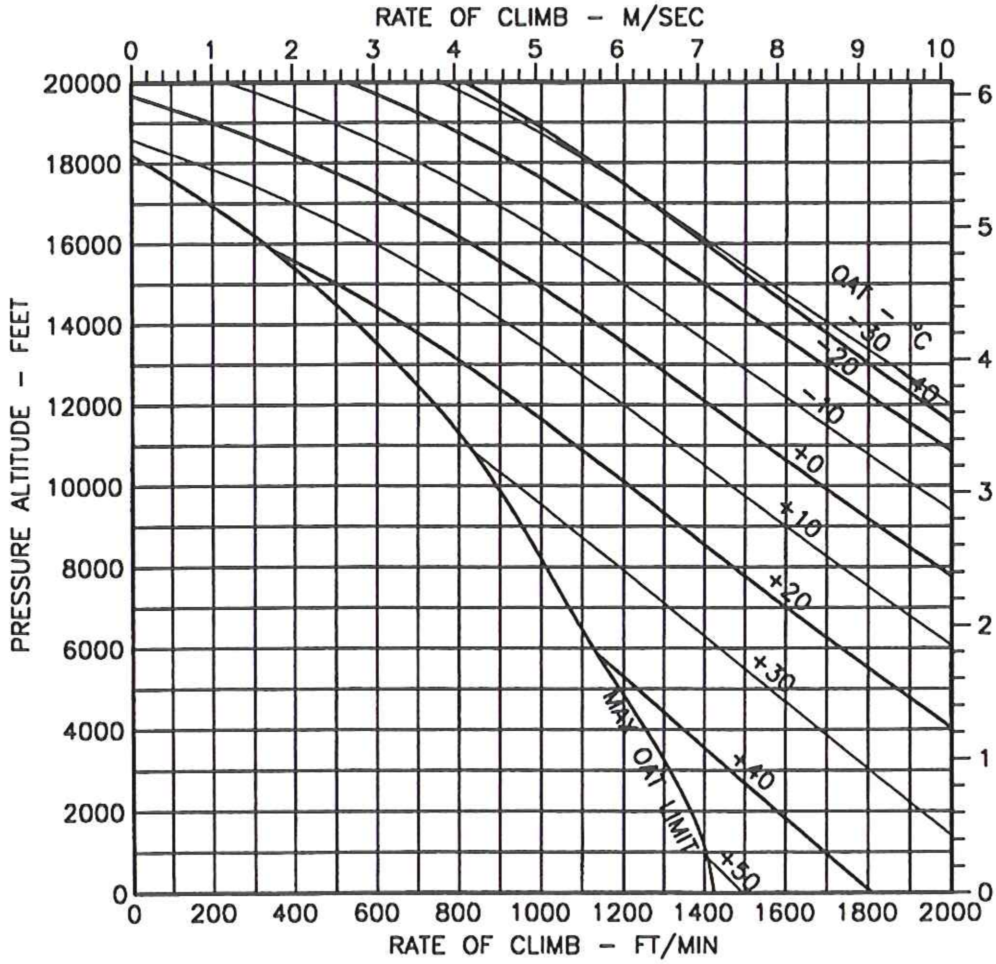

Figure 4-8. Rate of climb - takeoff power (sheet 4 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 110 FT/MIN ABOVE
5000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

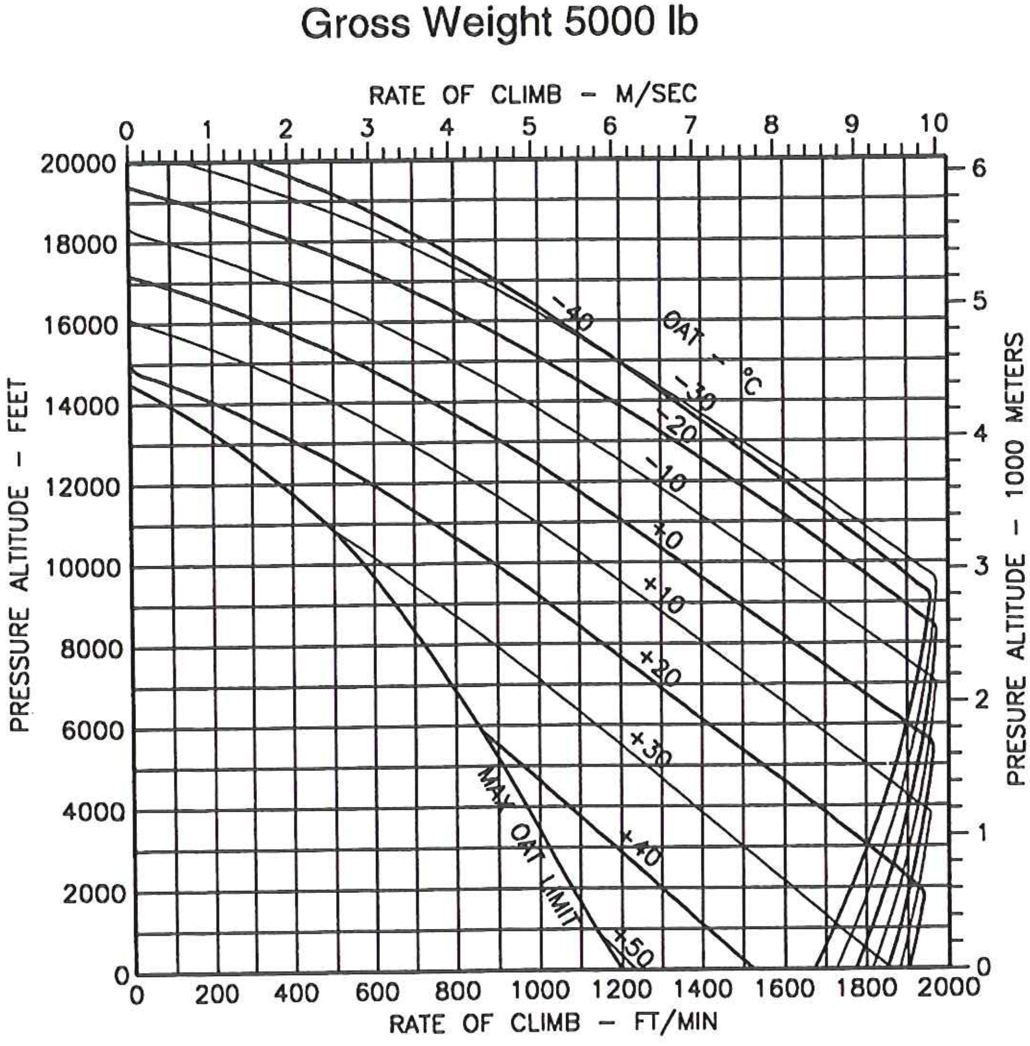

Figure 4-8. Rate of climb - takeoff power (sheet 5 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 180 FT/MIN ABOVE
14,500 FT H_P FOR ANTI-ICE ON (5°C AND COLDER)

Gross Weight 1400 kg

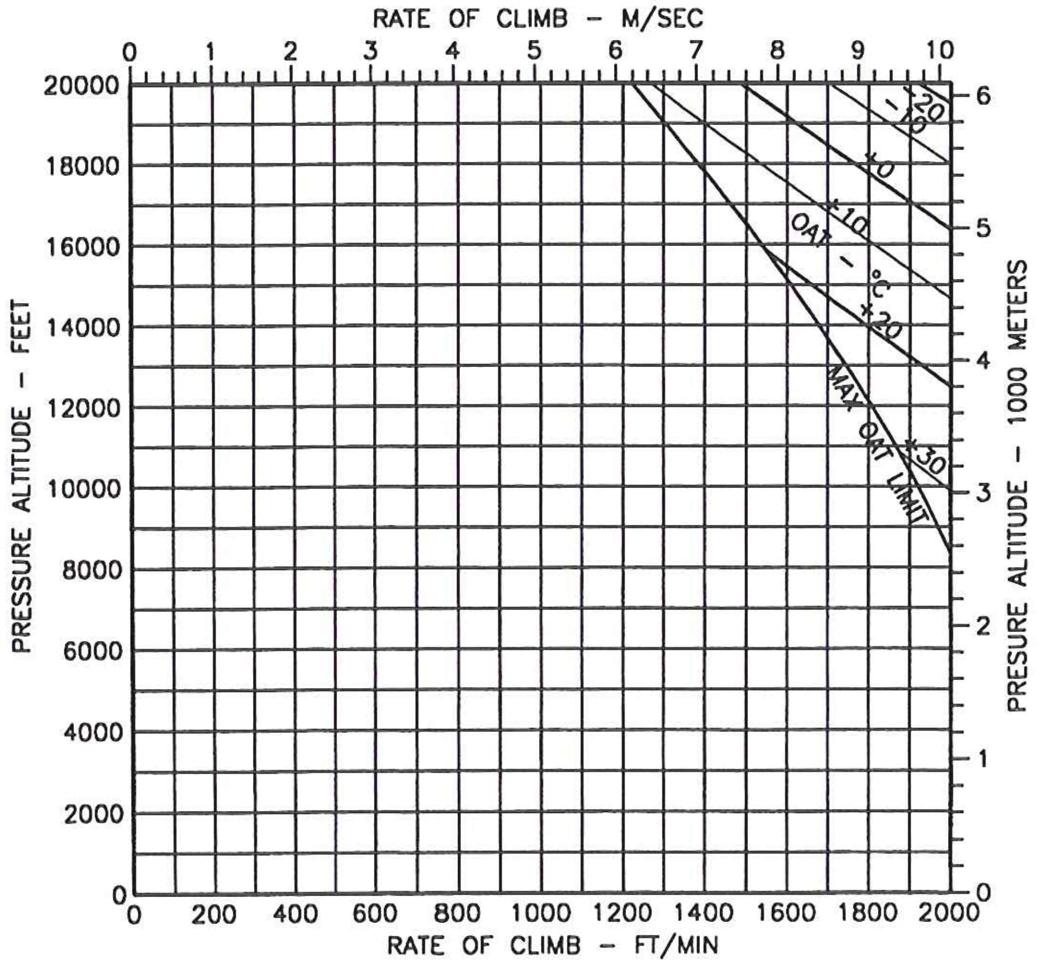

Figure 4-8. Rate of climb - takeoff power (sheet 6 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 155 FT/MIN ABOVE
11,500 FT H_P FOR ANTI-ICE ON (5°C AND COLDER)

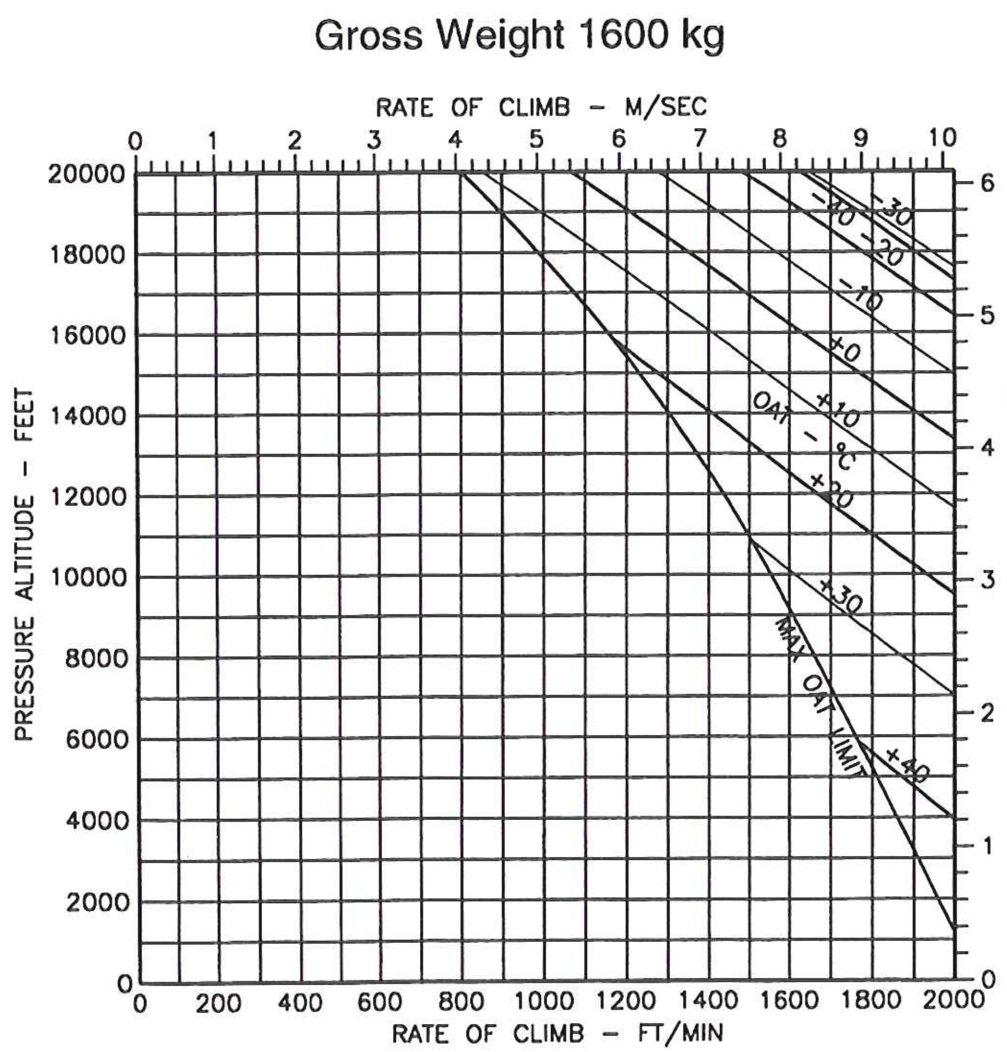

Figure 4-8. Rate of climb - takeoff power (sheet 7 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 135 FT/MIN ABOVE
9000 FT H_P FOR ANTI-ICE ON (5°C AND COLDER)

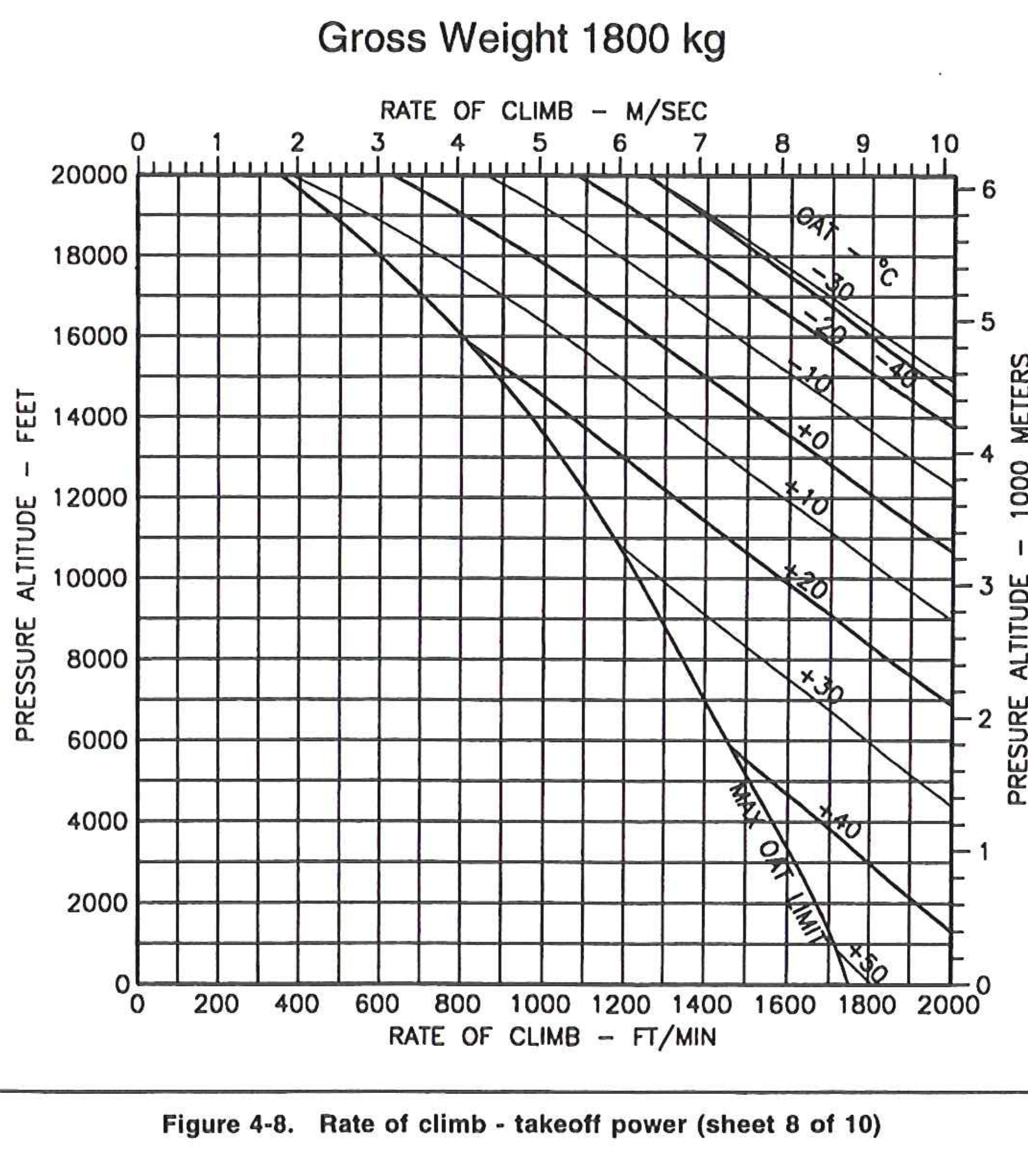

Figure 4-8. Rate of climb - takeoff power (sheet 8 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 120 FT/MIN ABOVE
7000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

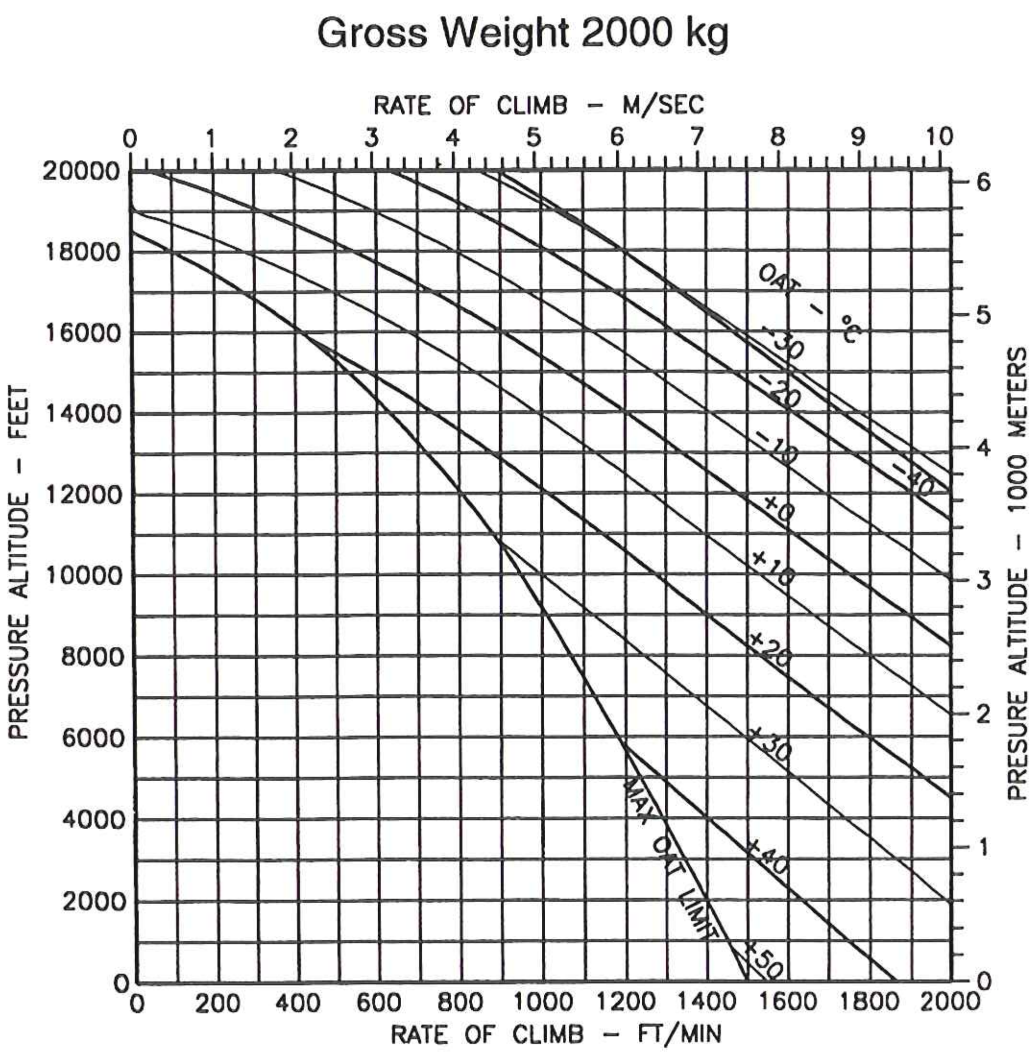

Figure 4-8. Rate of climb - takeoff power (sheet 9 of 10)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 110 FT/MIN ABOVE
5000 FT H_P FOR ANTI-ICE ON (5°C AND COLDER)

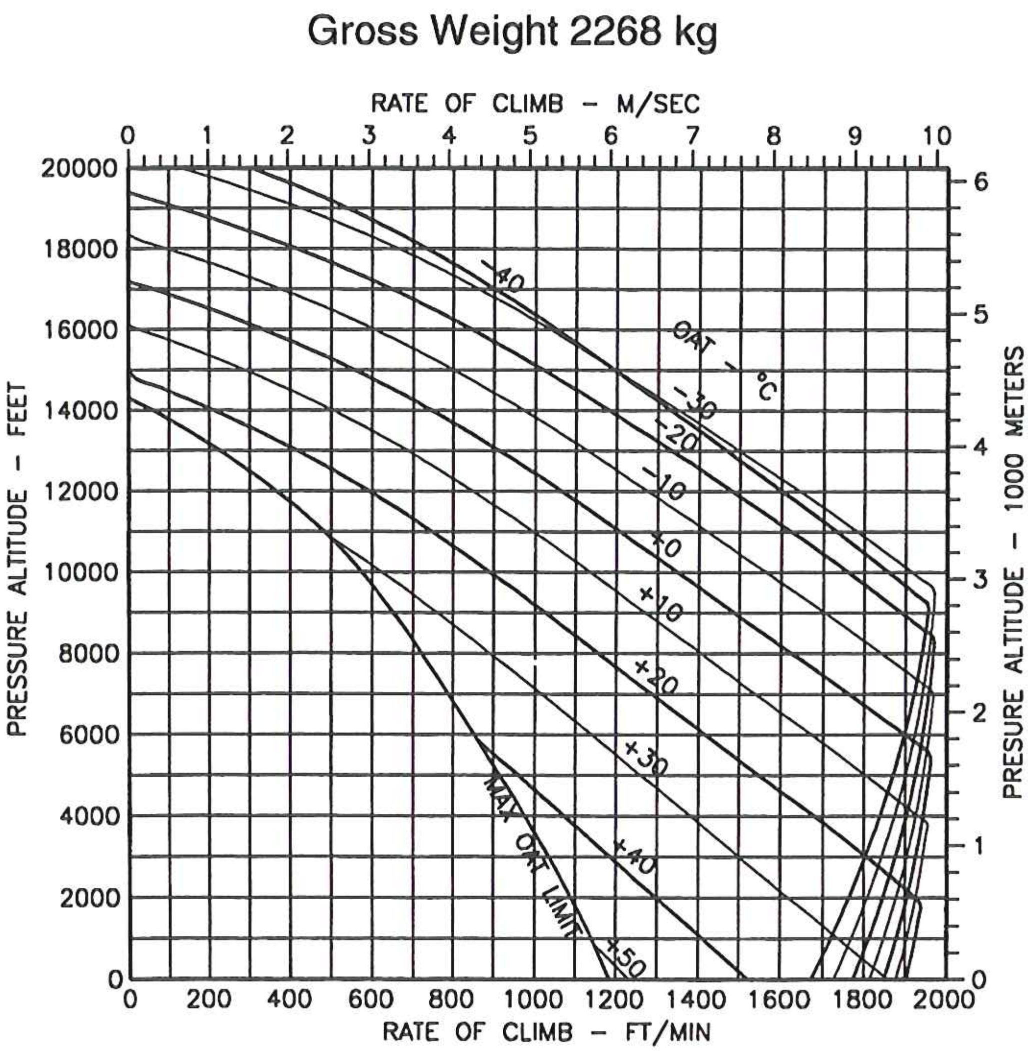

Figure 4-8. Rate of climb - takeoff power (sheet 10 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 225 FT/MIN ABOVE
10,000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

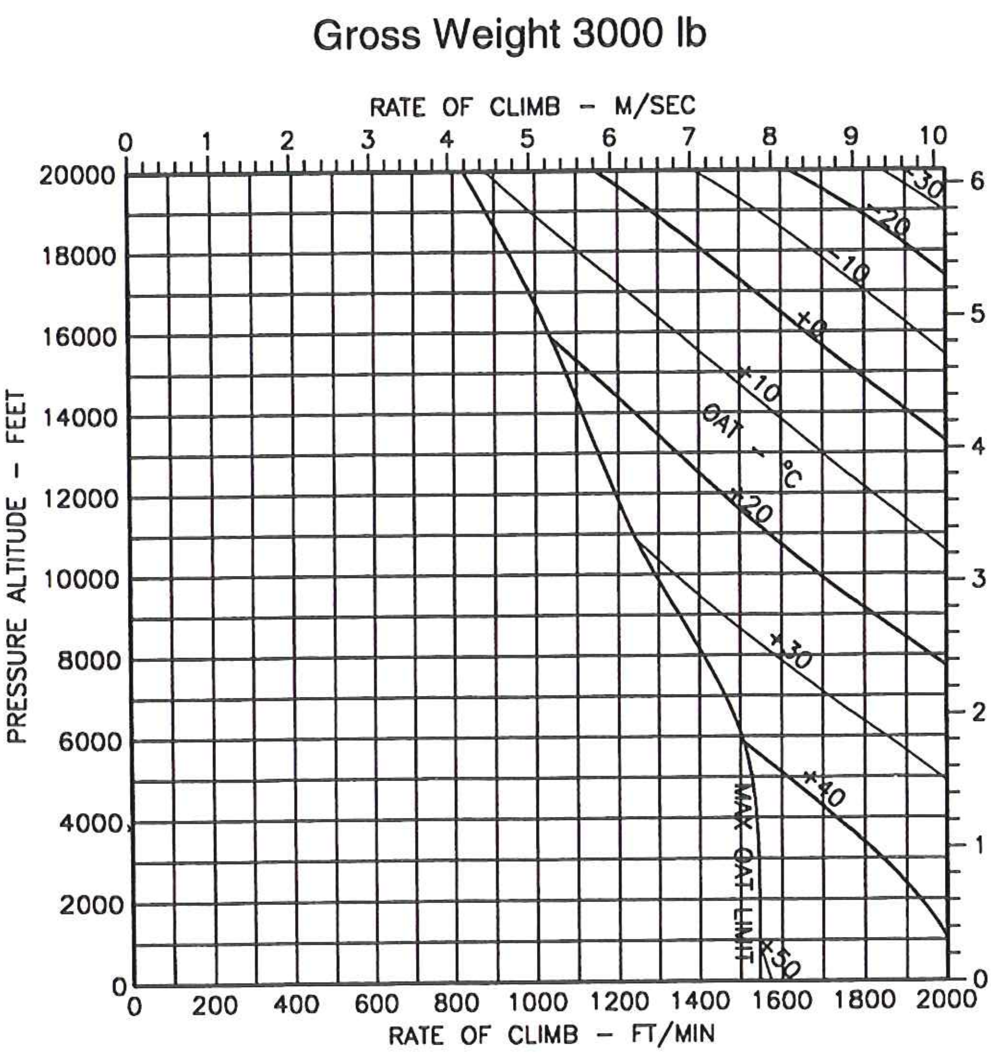

Figure 4-9. Rate of climb - maximum continuous power (sheet 1 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 190 FT/MIN ABOVE
7500 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

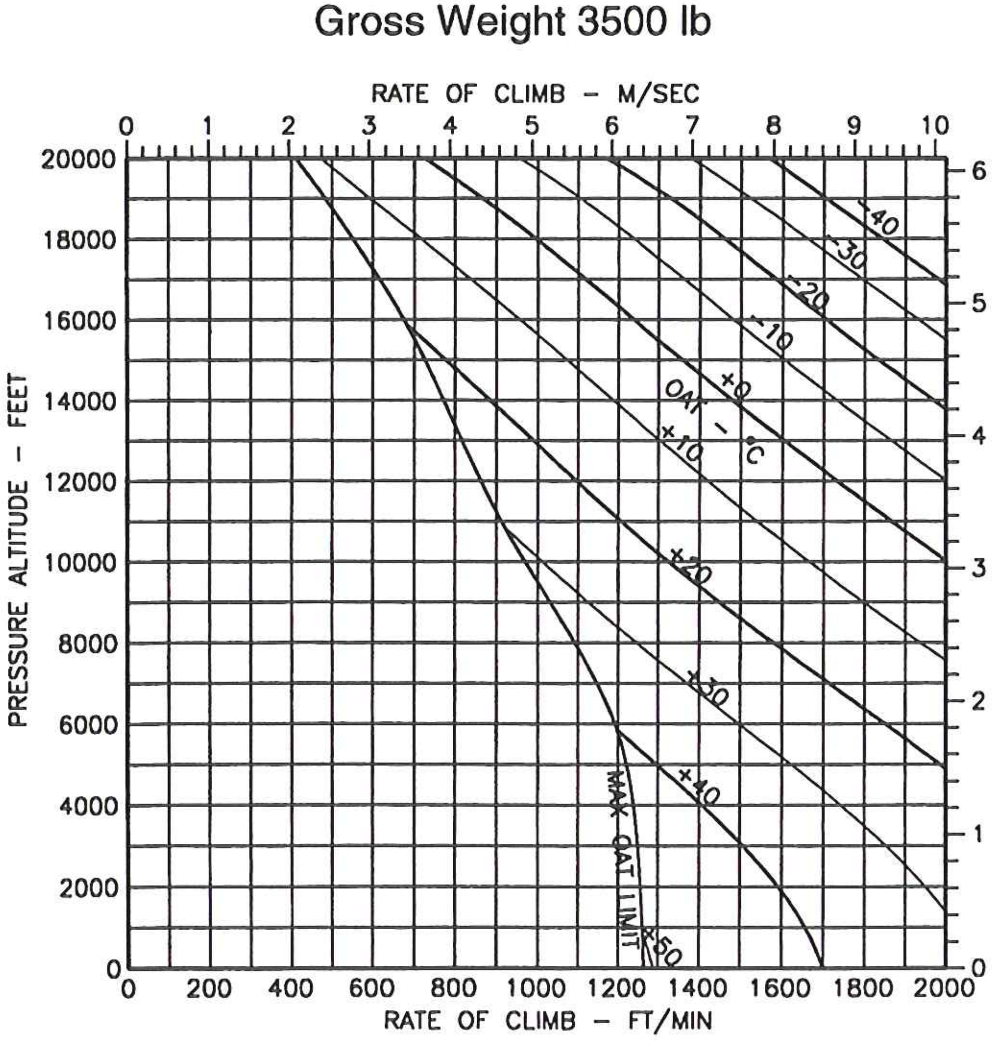

Figure 4-9. Rate of climb - maximum continuous power (sheet 2 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 170 FT/MIN ABOVE 5000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

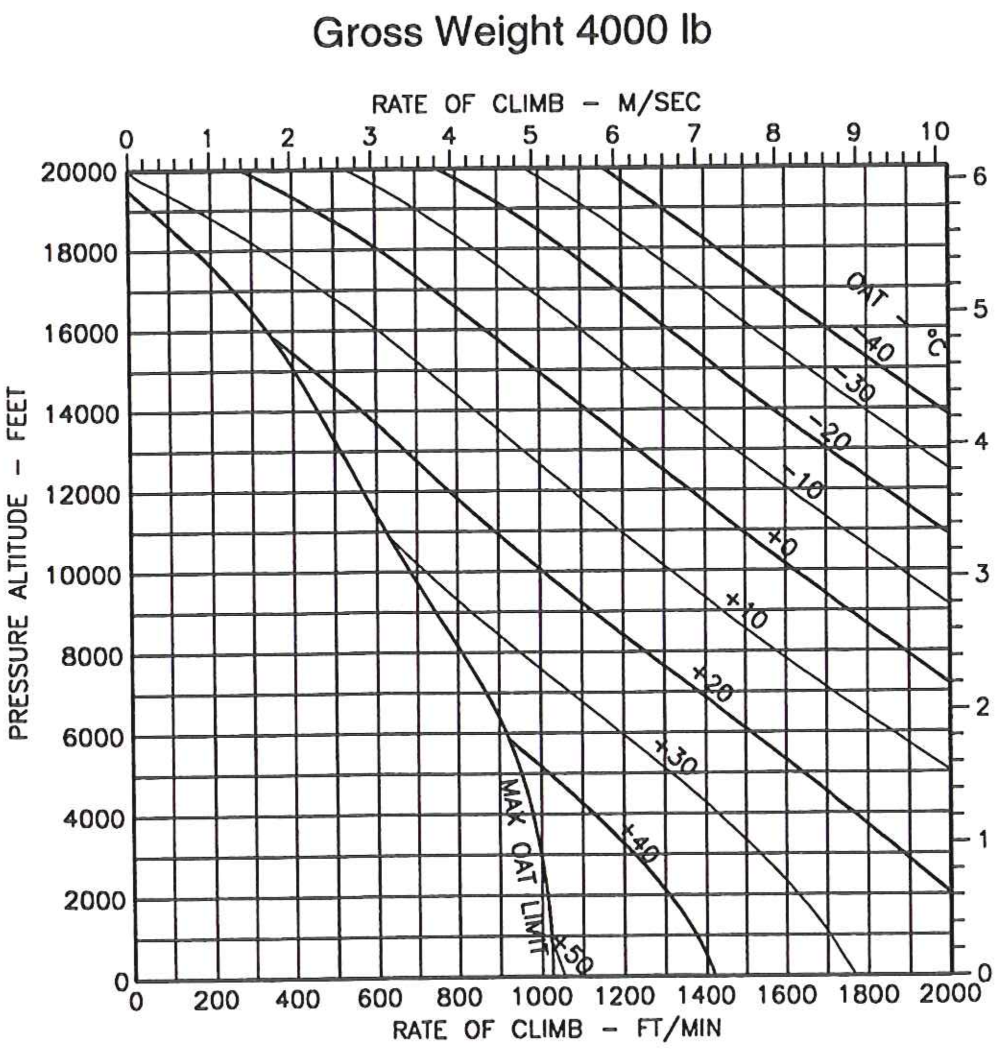

Figure 4-9. Rate of climb - maximum continuous power (sheet 3 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 150 FT/MIN ABOVE
3000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

Gross Weight 4500 lb

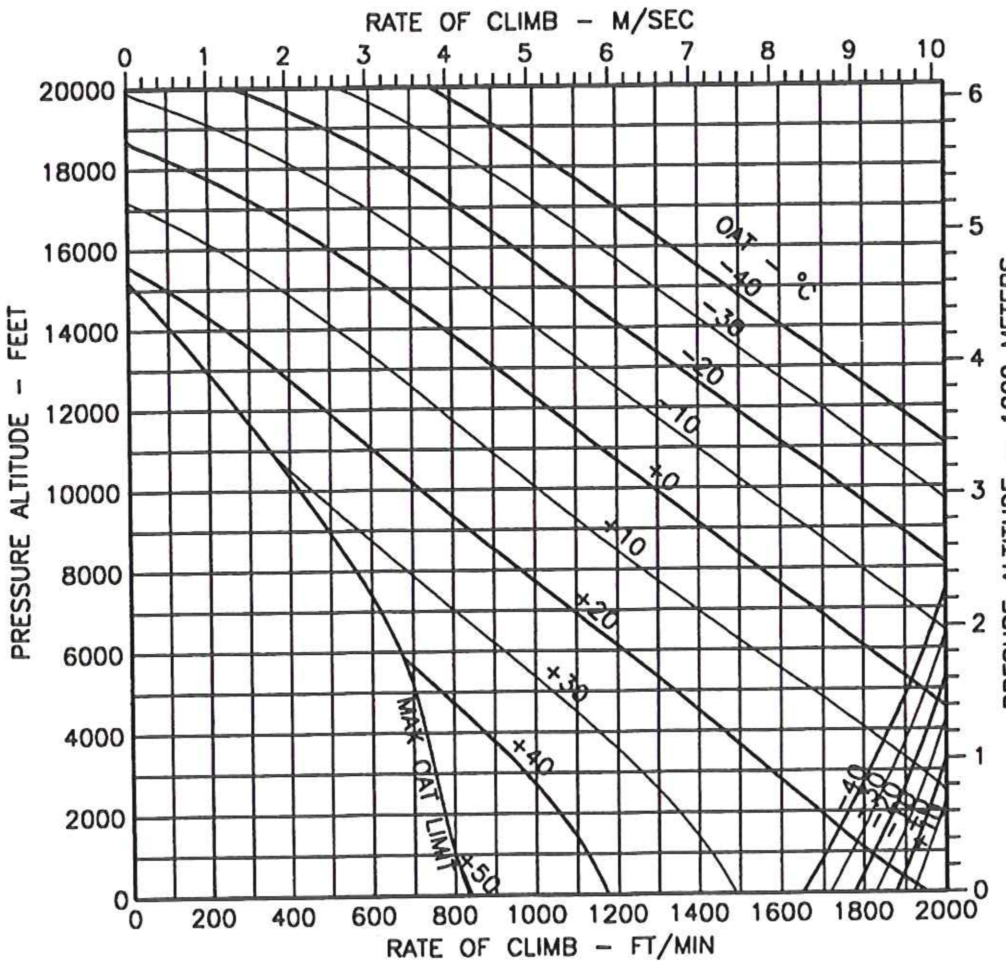

Figure 4-9. Rate of climb - maximum continuous power (sheet 4 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 135 FT/MIN ABOVE
3000 FT H_P FOR ANTI-ICE ON (5°C AND COLDER)

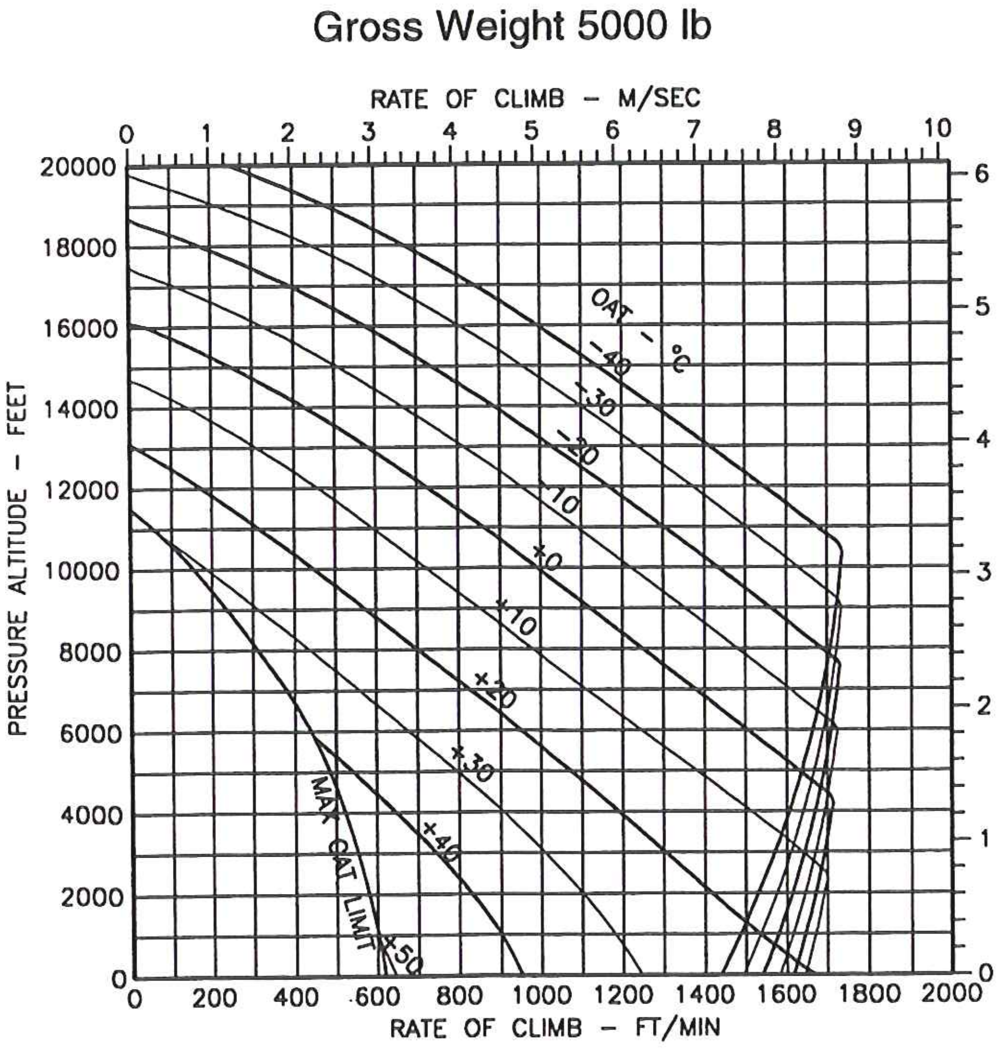

Figure 4-9. Rate of climb - maximum continuous power (sheet 5 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 225 FT/MIN ABOVE
9500 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

Gross Weight 1400 kg

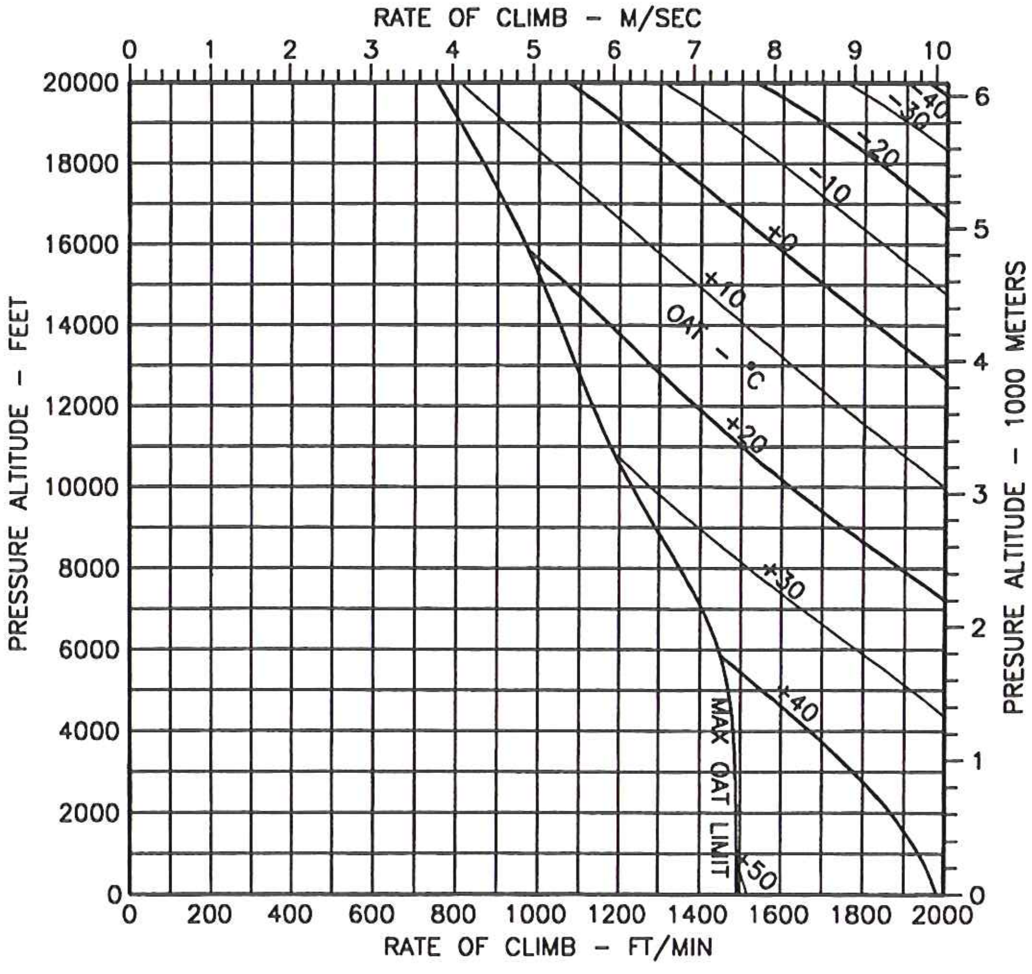

Figure 4-9. Rate of climb - maximum continuous power (sheet 6 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 190 FT/MIN ABOVE
7500 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

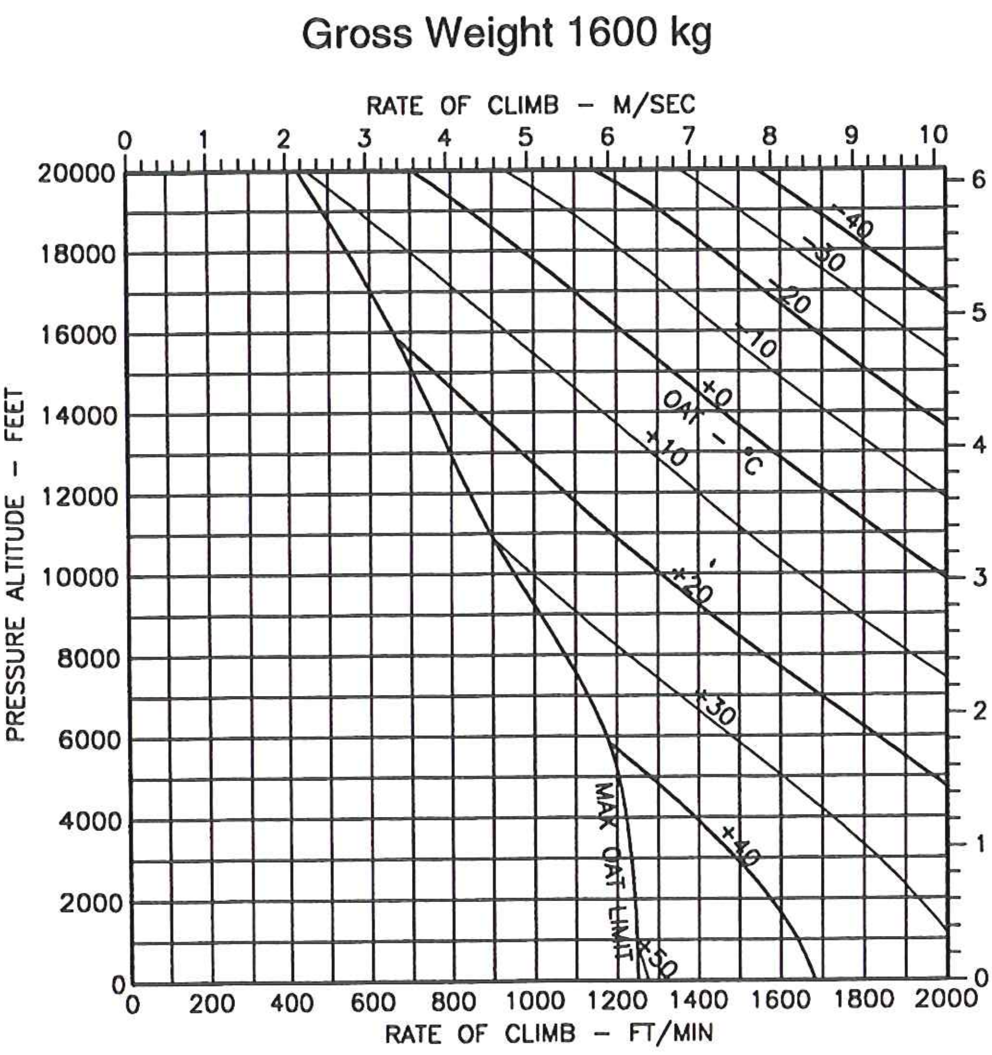

Figure 4-9. Rate of climb - maximum continuous power (sheet 7 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 170 FT/MIN ABOVE
5000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

Gross Weight 1800 kg

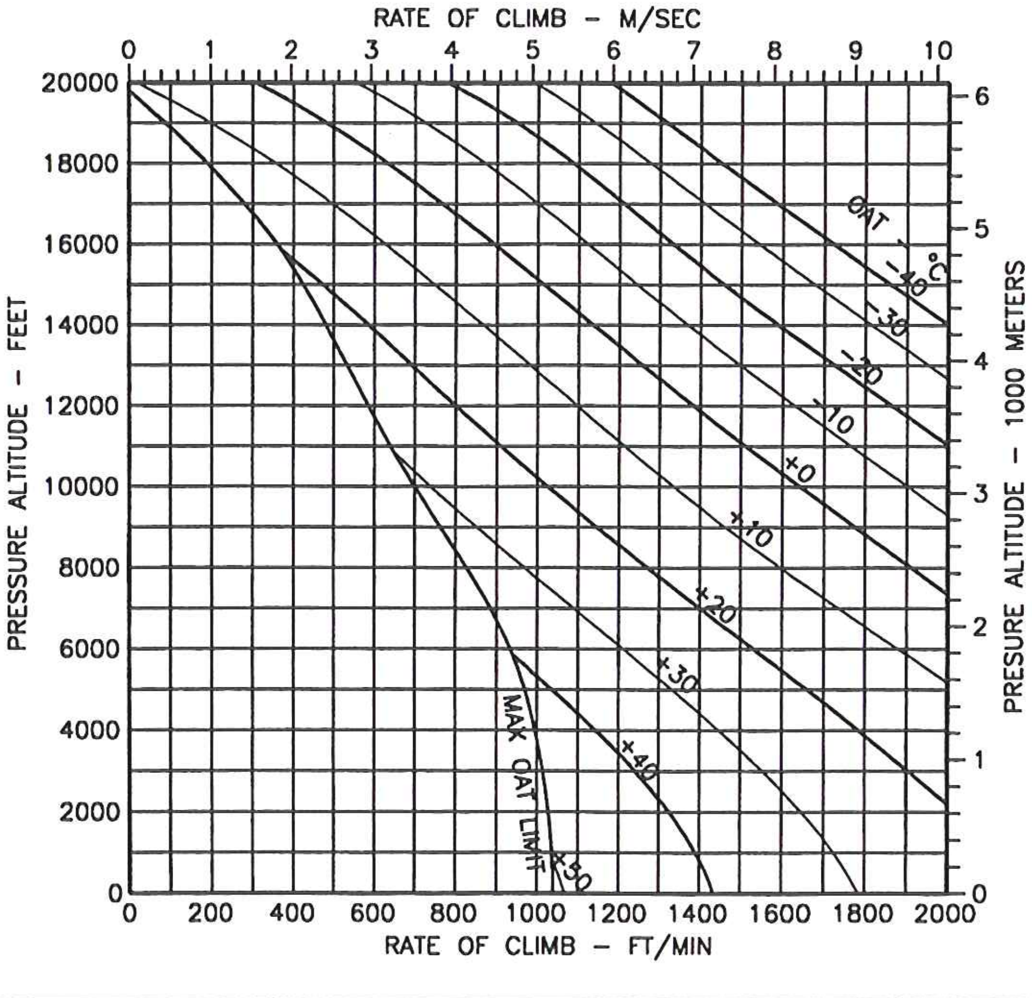

Figure 4-9. Rate of climb - maximum continuous power (sheet 8 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 150 FT/MIN ABOVE
3000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

Gross Weight 2000 kg

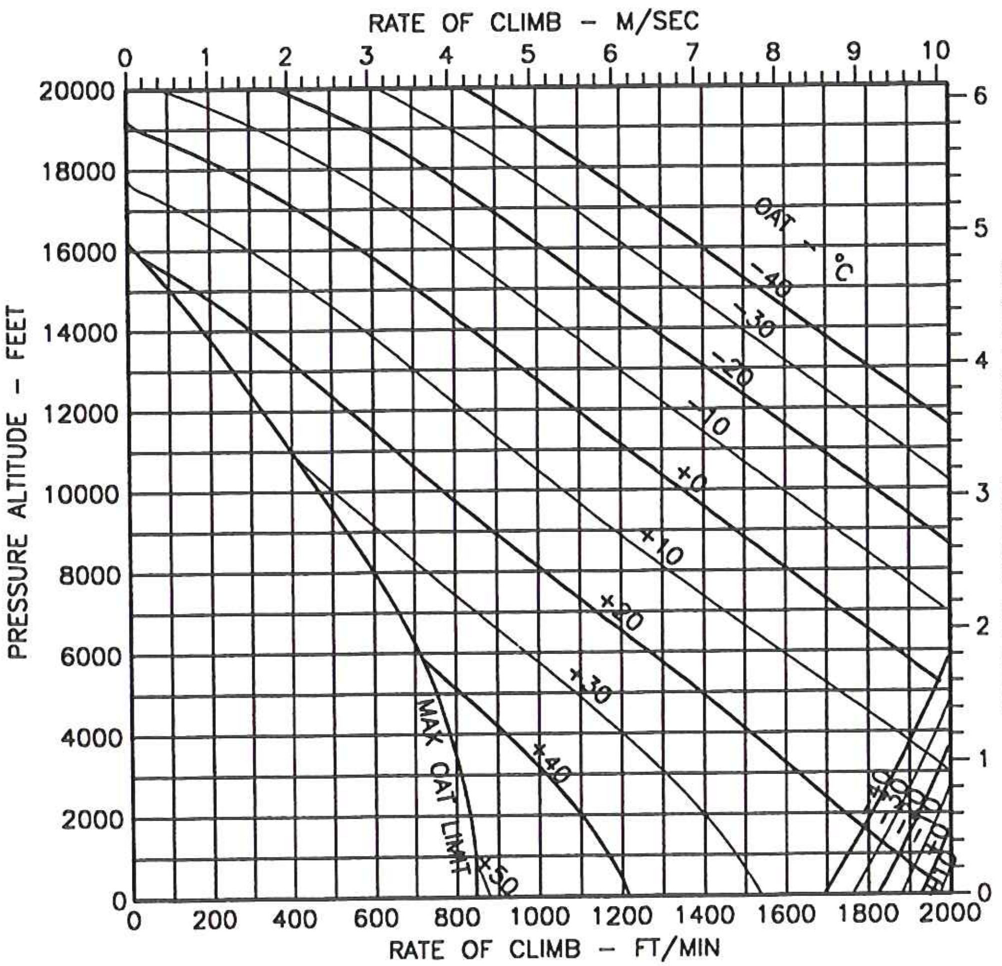

Figure 4-9. Rate of climb - maximum continuous power (sheet 9 of 10)

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RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER OFF

REDUCE RATE OF CLIMB 135 FT/MIN ABOVE
3000 FT H_p FOR ANTI-ICE ON (5°C AND COLDER)

Gross Weight 2268 kg

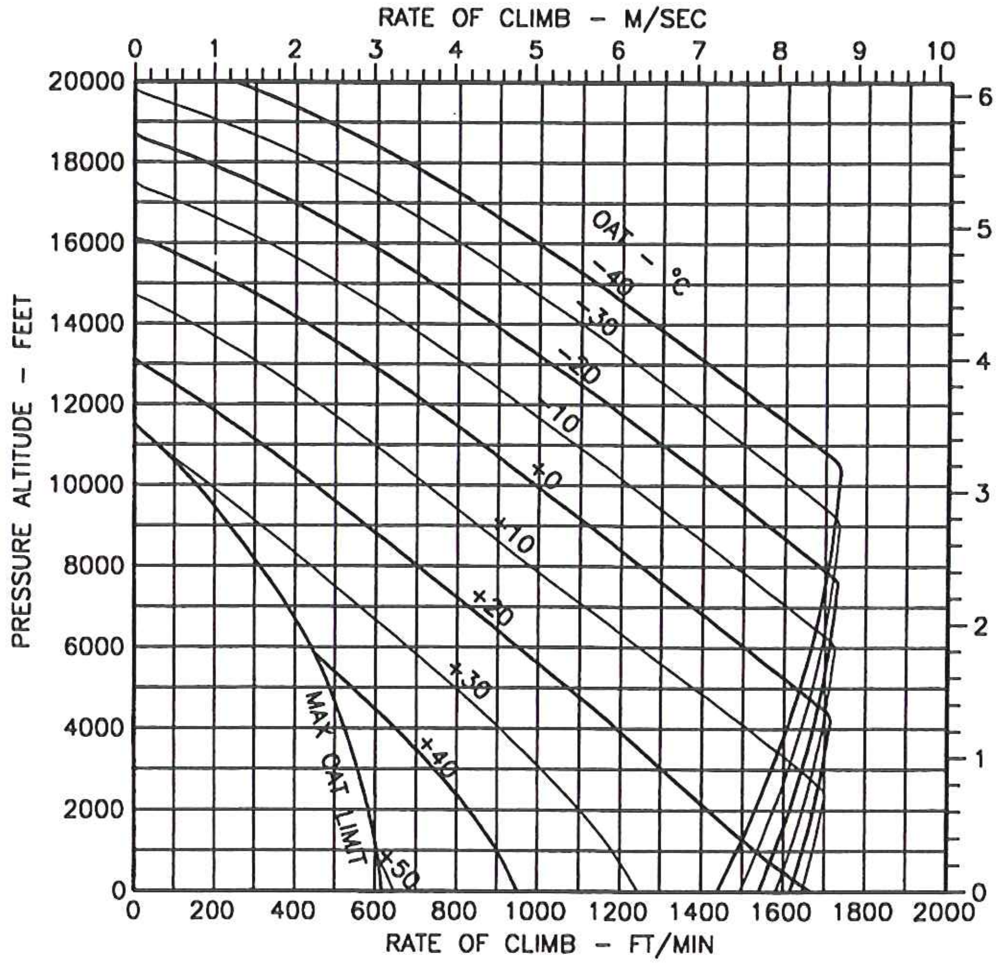

Figure 4-9. Rate of climb - maximum continuous power (sheet 10 of 10)

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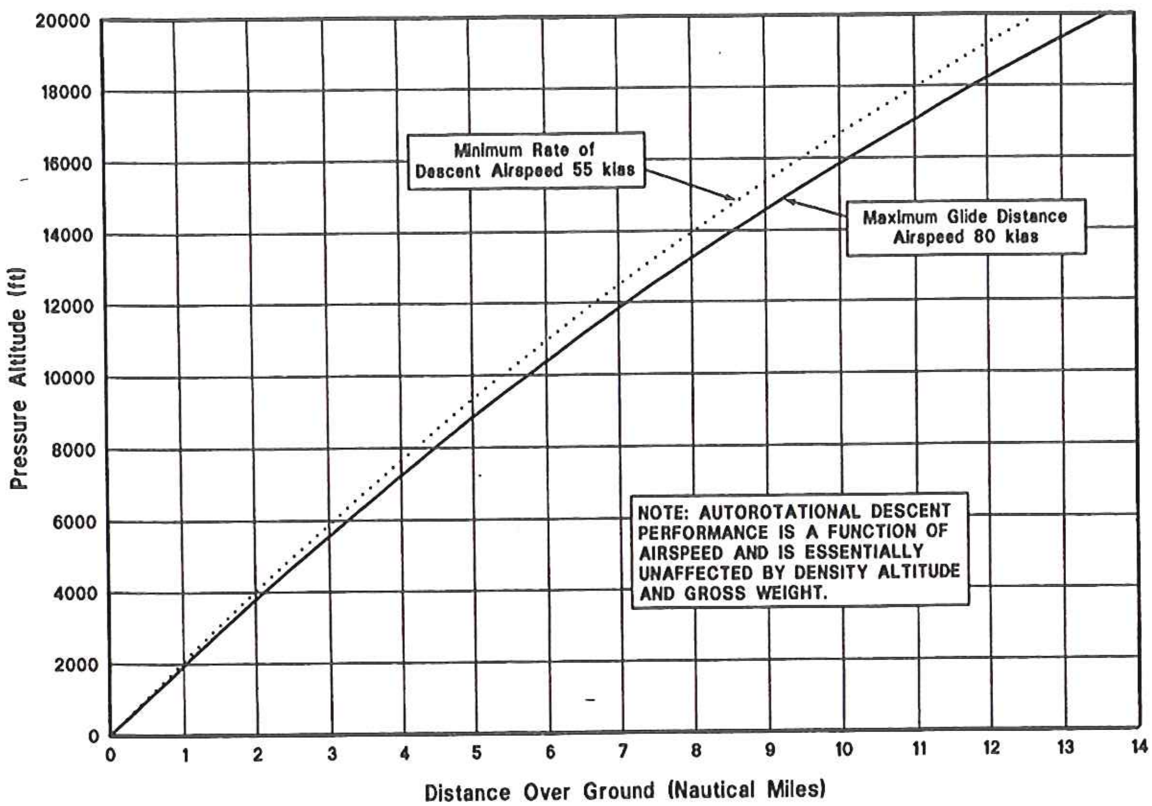

407FM-4-10

Figure 4-10. Autorotation glide distance

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AIRSPEED INSTALLATION CORRECTION TABLE
KCAS=(KIAS-INSTRUMENT ERROR-POSITION ERROR)
NOTE: This chart assumes zero instrument error.

KIAS	CLIMB KCAS	LEVEL FLIGHT KCAS
20	--	22
30	30	33
40	37	43
50	47	52
60	58	63
70	69	73
80	78	82
90	87	92
100	95	100
110	--	110
120	--	121
130	--	131
140	--	144

407-FM-4-11

Figure 4-11. Airspeed installation correction

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Section 5

WEIGHT AND BALANCE

5-1. INTRODUCTION

This section presents loading information and instructions necessary to ensure that flight can be performed within approved gross weight and center of gravity limitations as defined in Section 1.

5-2. EMPTY WEIGHT CENTER OF GRAVITY

5-2-A. EMPTY WEIGHT

The empty weight condition consists of the basic helicopter with required equipment, optional equipment kits, transmission and gearbox oils, hydraulic fluid, unusable fuel, undrainable engine oil, and fixed ballast. The empty weight and center of gravity are recorded on the Actual Weight Record, a copy of which should be carried in the helicopter to enable weight and balance computations.

5-2-B. CENTER OF GRAVITY

A empty weight center of gravity chart is provided in maintenance manual as a guide to simplify computing ballast requirements. This chart was derived from gross weight longitudinal center of gravity limits shown in Section 1, using most forward and most aft useful loads for standard seating and fuel.

NOTE

Empty weight center of gravity chart is not valid if helicopter has a nonstandard fuel system or seating arrangement.

5-3. GROSS WEIGHT CENTER OF GRAVITY

Gross weight condition is empty weight condition plus useful load.

5-3-A. USEFUL LOADS

Useful load consists of usable fuel, engine oil, crew, passengers, baggage and cargo. Combinations of these items which have most adverse effect on helicopter center of gravity are known as most forward and most aft useful loads. Whenever cargo and/or baggage are carried, these useful loads may be different for each flight, and weight and balance must be computed to ensure gross weight and center of gravity will remain within limits throughout flight.

Standard most forward and most aft useful loads are combinations of fuel, crew and passenger loading only. These loads, in conjunction with empty weight center of gravity chart, allow passengers only (no baggage or other cargo) to be carried within appropriate weight limitations without computing center of gravity for each flight.

If helicopter has a nonstandard fuel system or seating arrangement, or is not ballasted in accordance with empty weight center of gravity chart in maintenance manual, pilot must determine weight and balance to ensure gross weight and center of gravity will remain within limits throughout each flight.

5-3-B. CENTER OF GRAVITY

It is the responsibility of the pilot to ensure that helicopter is properly loaded

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to maintain center of gravity throughout each flight within gross weight center of gravity limits shown in Section 1 or appropriate supplement. Gross weight longitudinal and lateral center of gravity can be calculated using Actual Weight Record, diagrams and loading tables in this section and loading tables in applicable flight manual supplements.

When carrying baggage, cargo or nonstandard loads, effects of fuel consumption and addition/deletion of passengers, baggage or cargo at intermediate points should be checked prior to flight.

Significant fuselage stations and buttock lines are shown in figures 5-1 and 5-2 to aid in weight and balance computations.

5-4. DOORS OPEN OR REMOVED

When one or more cabin doors are removed, helicopter may exceed gross weight center of gravity limits during flight. If using Weight empty center of gravity chart, refer to BHT-407-MM-1, a ballast adjustment to offset moment change is necessary (Table 5-1). Otherwise, gross weight center of gravity should be computed for each flight.

5-4-A. DOOR WEIGHTS AND MOMENTS

Following table presents weight and moment adjustments for cabin doors. Sign convention for buttlock lines used to compute lateral moments are:

1. Left is negative.
2. Right is positive.

ACTION	MOMENT CHANGE	
	LEFT DOOR	RIGHT DOOR
Remove	Positive (+)	Negative (-)
Install	Negative (-)	Positive (+)

Example:

When removing a left door only, subtract positive weight value and negative moment value shown in table. Net effect on helicopter is a reduction in weight and a shift in lateral CG to right (positive direction).

5-4-B. BALLAST ADJUSTMENT

Following check can be made to determine if a ballast adjustment is necessary after doors are removed or installed.

1. For helicopters without ballast or with nose ballast, apply weight and moment changes to most aft useful load condition to determine if an increase in nose ballast is required, or a reduction is allowed.
2. For helicopters with tail ballast, apply weight and moment changes to most forward useful load condition to determine if a reduction in tail ballast is allowed, or an increase is required.

NOTE

Ballast changes are performed by maintenance personnel. After any ballast change, Actual Weight Record must be revised to show new weight empty condition.

5-5. COCKPIT AND CABIN LOADING

Loading tables (Tables 5-2 and 5-3) provide weights and moments for each passenger location, litter patient and baggage compartment in both U.S. and metric units.

To find moments for weights in excess of those shown on tables, multiply weight by fuselage station at which center of gravity of the object is located. An alternate method is to calculate amount of weight in

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excess of maximum weight listed on table, then read moment for this excess weight from table and add it to moment for maximum weight shown on table. This will give desired moment for the object.

5-5-A. LONGITUDINAL LOADING

1. A minimum weight of 170 pounds (77.1 kilograms) is required in cockpit at fuselage station 65.0 when the empty weight center of gravity chart is used.

2. Passenger seating is unrestricted.

3. Cargo loading is restricted only by floor load limit. Refer to Section 1.

5-5-B. MOST FORWARD AND MOST AFT CG

When using empty weight center of gravity chart, following combinations of crew, fuel and passenger loading will have most extreme effects on longitudinal center of gravity, assuming standard weights for all crew and passengers.

1. Most forward CG will occur with forward and mid seats occupied and fuel quantity of 74.8 gallons (283.0 liters).

2. Most aft CG will occur with one forward seat occupied (pilot) and fuel quantity of 28.4 gallons (107.5 liters).

Since center of gravity of aft passengers is on aft limit, weight of passengers is not included in most aft useful load. However, when most aft center of gravity of a configuration is forward of aft limit, addition of aft passengers will shift center of gravity further aft, and should be included in computation.

5-5-C. ALTERNATE LOADING

Gross weight center of gravity chart must be used to determine cabin loading requirements under following conditions:

1. Whenever cargo and/or baggage are carried.

2. When actual passenger weights are used.

3. When seating arrangement and/or fuel system are non-standard.

4. When performing specialty missions, such as hoisting or rappelling.

5-5-D. CABIN FLOOR LOADING

Cabin floor is structurally designed for 75 pounds per square foot (3.7 kilograms per 100 square centimeters)

5-6. BAGGAGE COMPARTMENT LOADING

When weight is loaded into baggage compartment, the pilot is required to compute weight and balance, regardless of passenger loading.

Baggage compartment is structurally designed for 86 pounds per square foot (4.2 kilograms per 100 square centimeters) for a total weight of 250 pounds (113.4 kilograms).

Loading of baggage compartment should be from front to rear. Load shall be secured to tiedown fittings if shifting of load in flight could result in structural damage to baggage compartment or in gross weight center of gravity being exceeded.

If load is not secured, center of gravity must be computed with load in most adverse position.

5-7. FUEL LOADING

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Longitudinal center of gravity of fuel shifts as it is consumed (Figure 5-3). Extreme effects of fuel consumption on helicopter center of gravity for standard fuel system are as follows:

1. Critical fuel for computing most forward useful load is 74.8 gallons (283.0 liters).

2. Critical fuel for computing most aft useful load is 28.4 gallons (107.5 liters).

Fuel loading tables (Table 5-4 and 5-5) list usable fuel quantities, weight and moments in both U.S. and metric units.

Fuel density vs temperature (Table 5-6), is provided to calculate fuel weight variation for equivalent volumes of fuel caused by a change in temperature. For example weight of 127.8 gallons (full fuel) of JP-5 at -40°F is 913.8 pounds (414.5 kilograms) versus 869.0 pounds (394.1 kilograms) shown on Fuel loading chart (Tables 5-4 and 5-5).

5-8. SAMPLE LOADING PROBLEM

A sample loading problem showing derivation of critical gross weights and center of gravity locations for a typical mission is presented in U.S. and metric units (Tables 5-7 and 5-8). Method shown derives a gross weight with zero fuel for each load condition to be checked, then adds appropriate fuel weight and moment read directly from Fuel loading table. Center of gravity for each condition is calculated by dividing total moment by total weight.

Forms have been provided (Tables 5-9 and 5-10) in both U.S. and Metric Units, to aid in computing critical load conditions for a flight.

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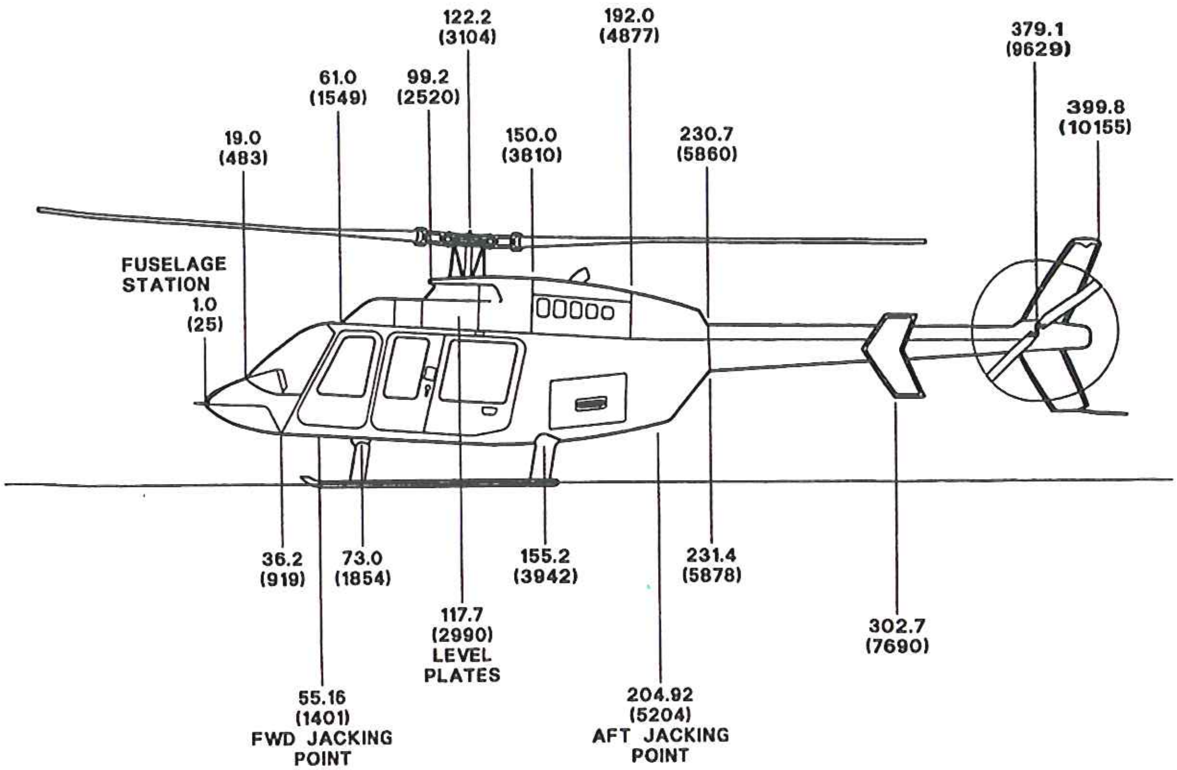

NOTE

Reference datum line, (Fuselage Station 0), is located 55.16 inches (1401 millimeters) forward of the forward jack point center line.

407L-MD-1-1

Figure 5-1. Fuselage stations

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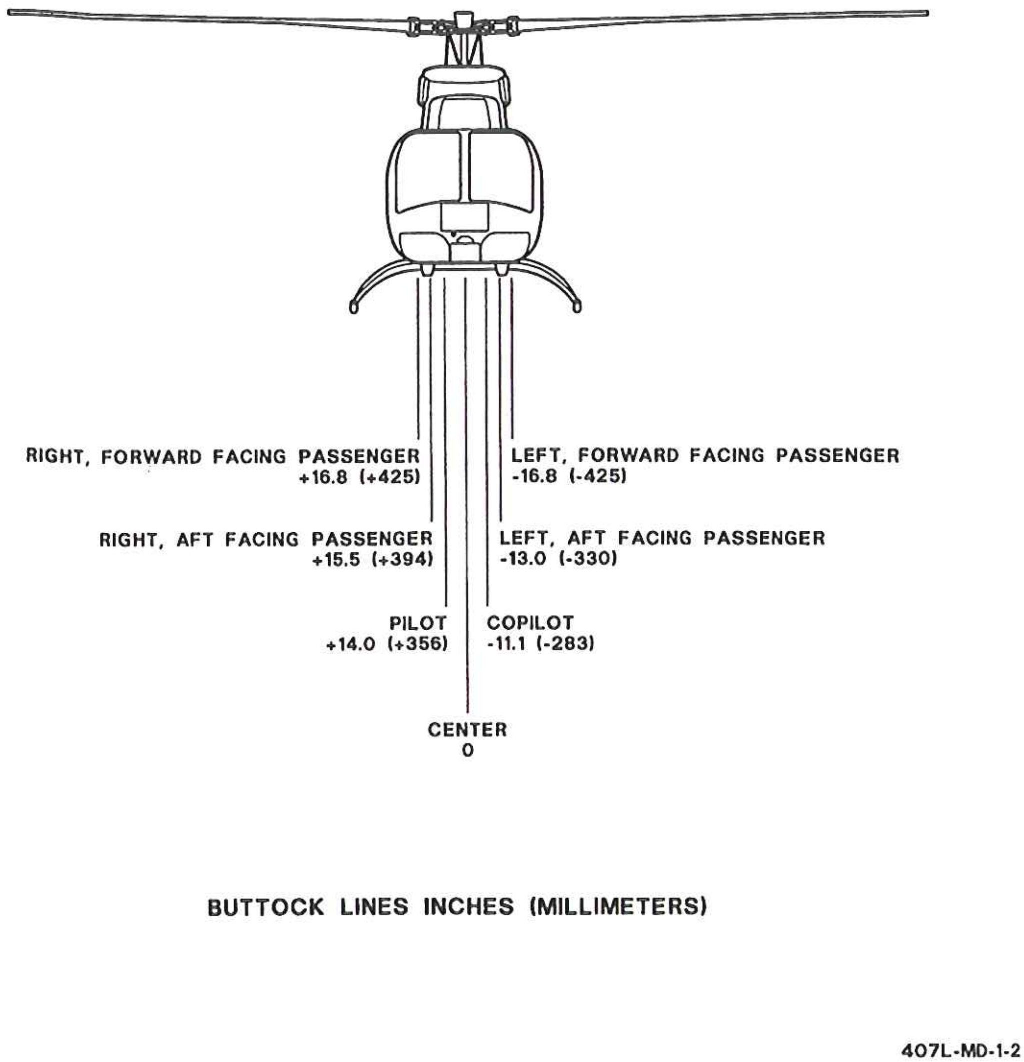

Figure 5-2. Buttock lines

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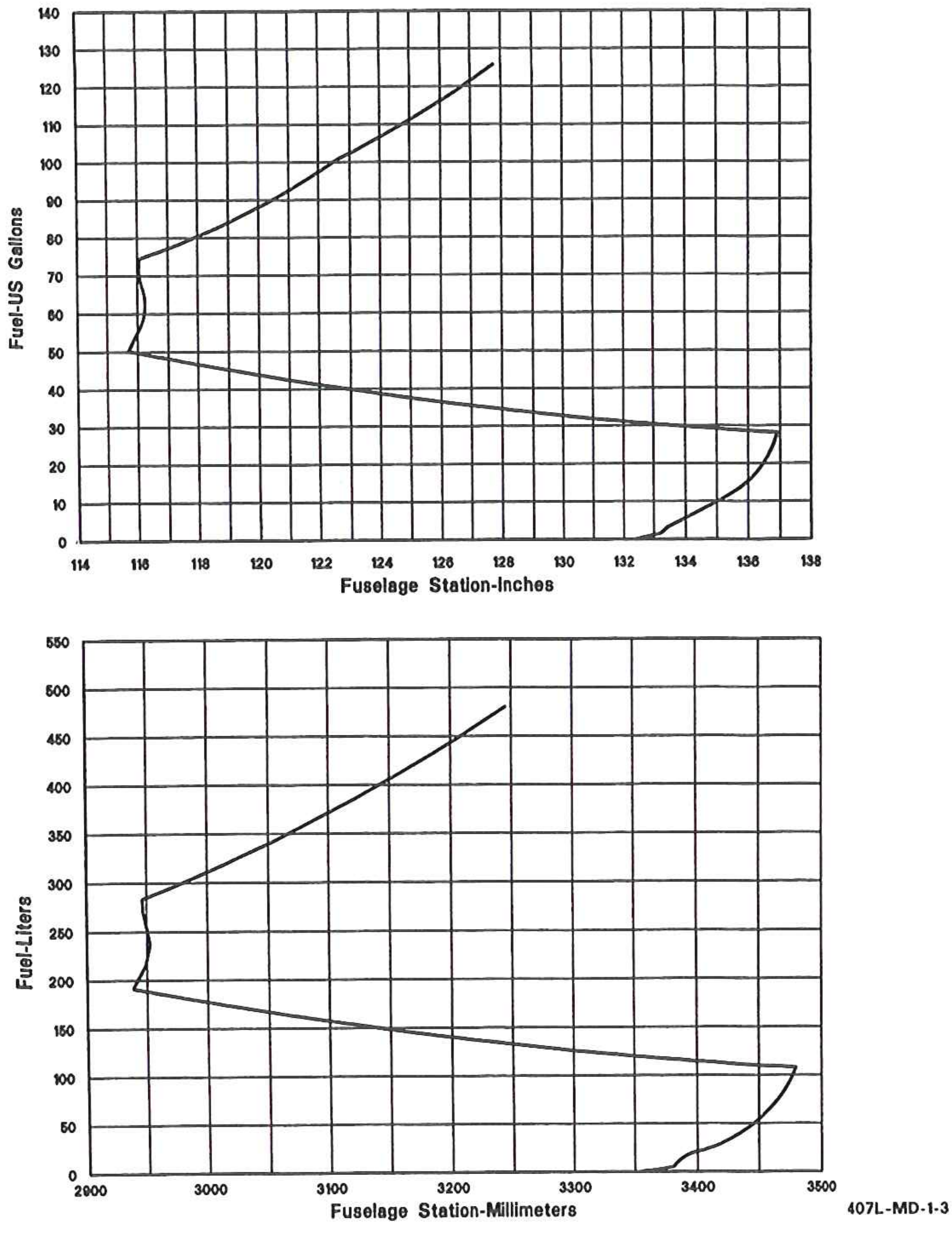

Figure 5-3. Fuel center of gravity

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Table 5-1. Door Weights and Moments (U.S.)

		LONGITUDINAL		LATERAL	
DOOR	WEIGHT (LB)	CG (IN)	MOMENT (IN-LBS)	CG (IN)	MOMENT (IN-LBS)
One crew door	13	64	832	±26	±338
Both crew doors	26	64	1664	0	0
One passenger door	15	125	1875	±27	±405
Both passenger doors	30	125	3750	0	0
Left passenger door and litter door	29	111	3219	–27	–783

Door Weights and Moments (Metric)

		LONGITUDINAL		LATERAL	
DOOR	WEIGHT (kg)	CG (mm)	MOMENT (kg•mm/100)	CG (mm)	MOMENT (kg•mm/100)
One crew door	5.9	1626	95.9	±660	±38.9
Both crew doors	11.8	1626	191.9	0	0
One passenger door	6.8	3175	215.9	±686	±46.6
Both passenger doors	13.6	3175	431.8	0	0
Left passenger door and litter door	13.2	2819	372.1	–586	–90.6

(TABLE I.D. 911618)

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Table 5-2. Cabin and baggage loading (U.S.)

CABIN AND BAGGAGE COMPARTMENT
TABLE OF MOMENTS
INCH-POUNDS

WEIGHT (LB)	FRONT SEAT FS 65	MID-PASS. (FACING AFT) FS 91	AFT-PASS. (FACING FWD) FS 129	LITTER PATIENT(S) FS 108	BAGGAGE FS 174
10	650	910	1290	1080	1740
20	1300	1820	2580	2160	3480
30	1950	2730	3870	3240	5220
40	2600	3640	5160	4320	6960
50	3250	4550	6450	5400	8700
60	3900	5460	7740	6480	10440
70	4550	6370	9030	7560	12180
80	5200	7280	10320	8640	13920
90	5850	8190	11610	9720	15660
100	6500	9100	12900	10800	17400
110	7150	10010	14190	11880	19140
120	7800	10920	15480	12960	20880
130	8450	11830	16770	14040	22620
140	9100	12740	18060	15120	24360
150	9750	13650	19350	16200	26100
160	10400	14560	20640	17280	27840
170	11050	15470	21930	18360	29580
180	11700	16380	23220	19440	31320
190	12350	17290	24510	20520	33060
200	13000	18200	25800	21600	34800
210	13650	19110	27090	22680	36540
220	14300	20020	28380	23760	38280
230	14950	20930	29670	24840	40020
240	15600	21840	30960	25920	41760
250	16250	22750	32250	27000	43500
260	16900	23660	33540	28080	
270	17550	24570	34830	29160	
280	18200	25480	36120	30240	
290	18850	26390	37410	31320	
300	19500	27300	38700	32400	
310	20150	28210	39990	33480	
320	20800	29120	41280	34560	
330	21450	30030	42570	35640	
340	22100	30940	43860	36720	
350	22750	31850	45150	37800	

(TABLE I.D. 911252)

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Table 5-3. Cabin and baggage loading (Metric)

CABIN AND BAGGAGE COMPARTMENT TABLE OF MOMENTS (mm – kg) / 100					
WEIGHT (kg)	FRONT SEAT 1651.0 mm	MID-PASS. (FACING AFT) 2311.4 mm	AFT-PASS. (FACING FWD) 3276.6 mm	LITTER PATIENT(S) 2743.2 mm	BAGGAGE 4419.6 mm
5	82.6	115.6	163.8	137.2	221.0
10	165.1	231.1	327.7	274.3	442.0
15	247.7	346.7	491.5	411.5	622.9
20	330.2	462.3	655.3	548.6	883.9
25	412.8	577.9	819.2	685.8	1104.9
30	495.3	693.4	983.0	923.0	1325.9
35	577.9	809.0	1146.8	960.1	1546.9
40	660.4	924.6	1310.6	1097.3	1767.8
45	743.0	1040.1	1474.5	1234.4	1988.8
50	825.5	1155.7	1638.3	1371.6	2209.8
55	908.1	1271.3	1802.1	1508.8	2430.8
60	990.6	1386.8	1966.0	1645.9	2651.8
65	1073.2	1502.4	2129.8	1783.1	2872.7
70	1155.7	1618.0	2293.6	1920.2	3093.7
75	1238.3	1733.6	2457.5	2057.4	3314.7
80	1320.8	1849.1	2621.3	2194.6	3535.7
85	1403.4	1964.7	2785.1	2331.7	3756.7
90	1485.9	2080.3	2948.9	2468.9	3977.6
95	1568.5	2195.8	3112.8	2606.0	4198.6
100	1651.0	2311.4	3276.6	2743.2	4419.6
105	1733.6	2427.0	3440.4	2880.4	4640.6
110	1816.1	2542.4	3604.3	3017.5	4861.6
113.4	1872.2	2621.1	3715.7	3110.8	5011.8
115	1898.7	2658.1	3768.1	3154.7	
120	1981.2	2773.7	3931.9	3291.8	
125	2063.8	2889.3	4095.8	3429.0	
130	2146.3	3004.8	4259.6	3566.2	
135	2228.9	3120.4	4423.4	3703.3	
140	2311.4	3236.0	4587.2	3840.5	
145	2394.0	3351.5	4751.1	3977.6	
150	2476.5	3467.1	4914.9	4114.8	

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Table 5-4. Fuel Loading (U.S.)

	JP-4	LONGITUDINAL			JP-5	LONGITUDINAL	
QUANTITY (U.S. GAL)	WEIGHT (LBS)	CG (IN)	MOMENT (IN-LBS)	QUANTITY (U.S. GAL)	WEIGHT (LBS)	CG (IN)	MOMENT (IN-LBS)
5	32.5	133.7	4,345	5	34.0	133.7	4,546
10	65.0	135.0	8,775	10	68.0	135.0	9,180
15	97.5	135.9	13,250	15	102.0	135.9	13,862
20	130.0	136.4	17,732	20	136.0	136.4	18,550
25	162.5	136.7	22,214	25	170.0	136.7	23,239
28.4△	184.6	137.0	25,290	28.4△	193.1	137.0	26,455
30	195.0	134.3	26,189	30	204.0	134.3	27,397
35	227.5	127.8	29,075	35	238.0	127.8	30,416
40	260.0	122.9	31,954	40	272.0	122.9	33,429
45	292.5	119.1	34,837	45	306.0	119.1	36,445
50	325.0	116.0	37,700	50	340.0	116.0	39,440
50.6••	328.9	115.7	38,054	50.6••	344.1	115.7	39,812
55	357.5	116.1	41,506	55	374.0	116.1	43,421
60	390.0	116.2	45,318	60	408.0	116.2	47,410
65	422.5	116.2	40,095	65	442.0	116.2	51,360
70	455.0	116.1	52,826	70	476.0	116.1	55,264
74.8☐	486.2	116.0	56,399	74.8☐	508.6	116.0	58,998
75	487.5	116.1	56,599	75	510.0	116.1	59,211
80	520.0	117.7	61,204	80	544.0	117.7	64,029
85	552.5	119.0	65,748	85	578.0	119.0	68,782
90	585.0	120.3	70,376	90	612.0	120.3	73,624
95	617.5	121.4	74,965	95	646.0	121.4	78,424
100	650.0	122.3	79,495	100	680.0	122.3	83,164
105	682.5	123.4	84,221	105	714.0	123.4	88,108
110	715.0	124.6	89,089	110	748.0	124.6	93,201
115	747.5	125.6	93,886	115	782.0	125.6	98,219
120	780.0	126.6	98,748	120	816.0	126.6	103,306
125	812.5	127.5	103,504	125	850.0	127.5	108,375
127.8★	830.7	127.9	106,247	127.8★	869.0	127.9	111,145

△ CRITICAL FUEL FOR MOST AFT C.G. CONDITION
•• MOST FORWARD FUEL C.G.
☐ CRITICAL FUEL FOR MOST FORWARD C.G. CONDITION)
★ FULL FUEL

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Table 5-5. Fuel Loading (Metric)

QUANTITY (LITERS)	JP-4 WEIGHT (kg)	LONGITUDINAL CG (mm)	LONGITUDINAL MOMENT (kg•mm/100)	QUANTITY (LITERS)	JP5,JP-8 WEIGHT (kg)	LONGITUDINAL CG (mm)	LONGITUDINAL MOMENT (kg•mm/100)
15	11.7	3389	397	15	12.2	3389	413
30	23.4	3415	799	30	24.4	3415	833
45	35.0	3439	1204	45	36.7	3439	1262
60	46.7	3455	1613	60	48.9	3455	1689
75	58.4	3465	2024	75	61.1	3465	2117
90	70.1	3472	2434	90	73.3	3472	2545
105	81.8	3478	2845	105	85.6	3478	2977
107.5Δ	83.7	3479	2912	107.5Δ	87.6	3479	3048
120	93.5	3352	3134	120	97.8	3352	3278
135	105.1	3228	3393	135	110.0	3228	3551
150	116.8	3129	3655	150	122.2	3129	3824
165	128.5	3049	3918	165	134.4	3049	4098
180	140.2	2982	4181	180	146.7	2982	4375
191.6••	149.2	2938	4383	191.6••	156.1	2938	4586
195	151.9	2940	4466	195	158.9	2940	4672
210	163.6	2949	4825	210	171.1	2949	5046
225	175.2	2951	5170	225	183.3	2951	5409
240	186.9	2953	5519	240	195.6	2953	5776
255	198.6	2950	5859	255	207.8	2950	6130
270	210.3	2948	6200	270	220.0	2948	6486
283.0☐	220.4	2948	6497	283.0☐	230.6	2948	6798
285	222.0	2951	6551	285	232.2	2951	6852
300	233.7	2983	6971	300	244.5	2983	7293
315	245.3	3012	7388	315	256.7	3012	7732
330	257.0	3038	7808	330	268.9	3038	8169
345	268.7	3061	8225	345	281.1	3061	8604
360	280.4	3083	8645	360	293.3	3083	9042
375	292.1	3103	9064	375	305.6	3103	9483
390	303.8	3123	9488	390	317.8	3123	9925
405	315.4	3147	9926	405	330.0	3147	10385
420	327.1	3169	10366	420	342.2	3169	10844
435	338.8	3190	10808	435	354.5	3190	11309
450	350.5	3210	11251	450	366.7	3210	11771
465	362.2	3228	11692	465	378.9	3228	12231
480	373.9	3245	12133	480	391.1	3245	12691
483.7★	376.7	3249	12239	483.7★	394.1	3249	12804

Δ CRITICAL FUEL FOR MOST AFT C.G. CONDITION
•• MOST FORWARD FUEL C.G.
☐ CRITICAL FUEL FOR MOST FORWARD C.G. CONDITION)
★ FULL FUEL

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Table 5-6. Fuel Density vs Temperature

TEMPERATURE DEG F	DENSITY LBS/GALLON	DENSITY LBS/GALLON	TEMPERATURE DEG C	DENSITY kg/liter	DENSITY kg/liter
	JP-4	JP-5		JP-4	JP-5
120	6.27	6.59	40	0.759	0.797
100	6.35	6.66	30	0.767	0.805
80	6.42	6.73	20	0.775	0.812
60*	6.50	6.80	15.56*	0.779	0.815
40	6.58	6.87	10	0.784	0.820
20	6.65	6.94	0	0.792	0.827
0	6.73	7.01	−10	0.800	0.835
−20	6.80	7.08	−20	0.808	0.842
−40	6.88	7.15	−30	0.816	0.850
			−40	0.824	0.857

*Standard density, used to derive fuel burn curves

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Table 5-7. Sample Loading Problem (U.S.)

A helicopter is chartered to transport 4 passengers plus pilot and 200 pounds of baggage on a trip that will require approximately 113 gallons of JP-5 fuel (one way). The pilot will return alone. Compute weight and center of gravity at takeoff and landing, and determine extreme cg conditions for both flights.

OUTBOUND FLIGHT

	Weight (Lbs)	Longitude CG (In)	Longitude Moment (In-Lbs)	Latitude CG (In)	Latitude Moment (In-Lbs)
Weight Empty	★2824.1	131.0	369957	0.1	317
+Oil	13.0	205.0	2665	0.0	0
+Pilot	200.0	65.0	13000	14.0	2800
+Forward Passenger	200.0	65.0	13000	−11.1	−2220
+Mid Passenger (1)	180.0	91.0	16380	15.5	2790
+Aft Passenger (2)	320.0	129.0	41280	0.0	0
+Baggage	200.0	174.0	34800	0.0	0
Gross Weight at Zero Fuel	3937.1	124.7	491082	0.9	3687
+Full Fuel (JP-5)	869.0	127.9	111145	0.0	0
Takeoff Gross Weight	4806.1√	125.3√	602227	0.8√	3687
Gross Weight at Zero Fuel	3937.1	124.7	491082	0.9	3687
+Critical Fuel for Most Forward	508.6	116.0	58998	0.0	0
Most Forward CG Condition	4445.7√	123.7√	550080	0.8√	3687
Gross Weight at Zero Fuel	3937.1	124.7	491082	0.9	3687
+Critical Fuel for Most Aft	193.1	137.0	26455	0.0	0
Most Aft CG Condition	4130.2√	125.3√	517537	0.9√	3687
Gross Weight at Zero Fuel	3937.1	124.7	491082	0.9	3687
+Fuel Remaining at Landing (14.8 gal)	100.6	135.9	13672	0.0	0
Landing Condition	4037.7√	125.0√	504754	0.9√	3687

RETURN FLIGHT

	Weight (Lbs)	Longitude CG (In)	Longitude Moment (In-Lbs)	Latitude CG (In)	Latitude Moment (In-Lbs)
Weight Empty	★2824.1	131.0	369957	0.1	317
+Oil	13.0	205.0	2665	0.0	0
+Pilot	200.0	65.0	13000	14.0	2800
Gross Weight at Zero Fuel	3037.1	127.0	385622	1.0	3117
+Full Fuel (JP-5)	869.0	127.9	111145	0.0	0
Takeoff Gross Weight	3906.1√	127.2√	496767	0.8√	3117
Gross Weight at Zero Fuel	3037.1	127.0	385622	1.0	3117
+Critical Fuel for Most Forward	508.6	116.0	58998	0.0	0
Most Forward CG Condition	3545.7√	125.4√	444620	0.9√	3117
Gross Weight at Zero Fuel	3037.1	127.0	385622	1.0	3117
+Critical Fuel for Most Aft	193.1	137.0	26455	0.0	0
Most Aft CG Condition	3230.2√	127.6√	412077	1.0√	3117
Gross Weight at Zero Fuel	3037.1	127.0	385622	1.0	3117
+Fuel Remaining at Landing (14.8 gal)	100.6	135.9	13672	0.0	0
Landing Condition	3137.7√	127.3√	399294	1.0√	3117

★ Example only. Refer to Actual Weight Record for actual Weight Empty data.

√ A check of the weight and cg values against the gross weight center of gravity limits chart shows that the loading will be within limits throughout flight. In lateral calculations, − is left side and + is right side.

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Table 5-8. Sample Loading Problem (Metric)

A helicopter is chartered to transport 4 passengers and 90.7 kilograms of baggage on a trip that will require approximately 427 liters of JP-5 fuel (one way). The pilot will return alone. Compute weight and center of gravity at takeoff and landing, and determine extreme cg conditions for both flights.

OUTBOUND FLIGHT

		LONGITUDE		LATITUDE	
	WEIGHT (kg)	CG (mm)	MOMENT (kg•mm/100)	CG (mm)	MOMENT (kg•mm/100)
Weight Empty	★1281.0	3327	42618.9	3	36.7
+Oil	5.9	5207	307.2	0	0.0
+Pilot	90.7	1651	1497.5	356	322.9
+Forward Passenger	90.7	1651	1497.5	−283	−256.7
+Mid Passenger (1)	81.6	2311	1885.8	394	321.5
+Aft Passenger (2)	145.2	3277	4758.2	0	0.0
+Baggage	90.7	4420	4008.9	0	0.0
Gross Weight at Zero Fuel	1785.8	3168	56573.9	24	424.4
+Full Fuel (JP-5)	394.1	3249	12804.3	0	0.0
Takeoff Gross Weight	2179.9√	3183√	69378.2	19√	424.4
Gross Weight at Zero Fuel	1785.8	3168	56573.9	24	424.4
+Critical Fuel for Most Forward	230.6	2948	6798.1	0	0.0
Most Forward CG Condition	2016.4√	3143√	63372.0	21√	424.4
Gross Weight at Zero Fuel	1785.8	3168	56573.9	24	424.4
+Critical Fuel for Most Aft	87.6	3479	3047.6	0	0.0
Most Aft CG Condition	1873.4√	3183√	59621.5	23√	424.4
Gross Weight at Zero Fuel	1785.8	3168	56573.9	24	424.4
+Fuel Remaining at Landing (56.7 liters)	46.2	3469	1602.7	0	0.0
Landing Condition	1832.0√	3176√	58176.6	23√	424.4

RETURN FLIGHT

Weight Empty	★1281.0	3327	42618.9	3	36.7
+Oil	5.9	5207	307.2	0	0.0
+Pilot	90.7	1651	1497.5	356	322.9
Gross Weight at Zero Fuel	1377.6	3225	44423.5	26	359.6
+Full Fuel (JP-5)	394.1	3249	12804.3	0	0.0
Takeoff Gross Weight	1771.7√	3230√	57227.8	20√	359.6
Gross Weight at Zero Fuel	1377.6	3225	44423.5	26	359.6
+Critical Fuel for Most Forward	230.6	2948	6798.1	0	0.0
Most Forward CG Condition	1608.2√	3185√	51221.6	22√	359.6
Gross Weight at Zero Fuel	1377.6	3225	44423.5	26	359.6
+Critical Fuel for Most Aft	87.6	3479	3047.6	0	0.0
Most Aft CG Condition	1465.2√	3240√	47471.1	25√	359.6
Gross Weight at Zero Fuel	1377.6	3225	44423.5	26	359.6
+Fuel Remaining at Landing (56.7 liters)	46.2	3469	1602.7	0	0.0
Landing Condition	1423.8√	3233√	46026.2	25√	359.6

★ Example only. Refer to Actual Weight Record for actual Weight Empty data.

√ A check of the weight and cg values against the gross weight center of gravity limits chart shows that the loading will be within limits throughout flight. In lateral calculations, – is left side and + is right side.

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Table 5-9. Weight and Balance Worksheet (U.S.)

WEIGHT AND BALANCE WORKSHEET (U.S.)					
		LONGITUDINAL		LATERAL	
	WEIGHT (LBS)	ARM (IN)	MOMENT (IN-LBS)	ARM (IN)	MOMENT (IN-LBS)
Weight Empty					
+Oil	13.0	205.0	2665	0.0	0
+Pilot		65.0		14.0	
+Forward Passenger		65.0		−11.1	
+Mid Passenger (L)		91.0		−13.0	
+Mid Passenger (R)		91.0		15.5	
+Aft Passenger (L)		129.0		−16.8	
+Aft Passenger (M)		129.0		0.0	
+Aft Passenger (R)		129.0		16.8	
+Baggage					
+Litter					
Gross Weight at Zero Fuel					
+Fuel				0.0	0
Takeoff Gross Weight					
Gross Weight at Zero Fuel					
+Critical Fuel for Most Forward		116.0		0.0	0
Most Forward CG Condition					
Gross Weight at Zero Fuel					
+Critical Fuel for Most Aft		137.0		0.0	0
Most Aft CG Condition					
Gross Weight at Zero Fuel					
+Fuel Remaining at Landing				0.0	0
Landing CG Condition					

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Table 5-10. Weight and Balance Worksheet (Metric)

WEIGHT AND BALANCE WORKSHEET (METRIC)					
		LONGITUDINAL		LATERAL	
	WEIGHT (kg)	ARM (mm)	MOMENT (kg•mm/100)	ARM (mm)	MOMENT (kg•mm/100)
Weight Empty					
+Oil	5.9	5207	307.2	0	0.0
+Pilot		1651		356	
+Forward Passenger		1651		−283	
+Mid Passenger (L)		2311		−330	
+Mid Passenger (R)		2311		394	
+Aft Passenger (L)		3277		−425	
+Aft Passenger (M)		3277		0	
+Aft Passenger (R)		3277		425	
+Baggage					
+Litter					
Gross Weight at Zero Fuel					
+Fuel				0	0.0
Takeoff Gross Weight					
Gross Weight at Zero Fuel					
+Critical Fuel for Most Forward		2948		0	0.0
Most Forward CG Condition					
Gross Weight at Zero Fuel					
+Critical Fuel for Most Aft		3479		0	0.0
Most Aft CG Condition					
Gross Weight at Zero Fuel					
+Fuel Remaining at Landing				0	0.0
Landing Condition					

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BELL MODEL 407

ROTORCRAFT FLIGHT MANUAL

SUPPLEMENT

INCREASED INTERNAL GROSS WEIGHT

407-706-020

CERTIFIED
16 MARCH 1999

This supplement shall be attached to Model 407 Flight Manual when INCREASED INTERNAL GROSS WEIGHT kit is installed.

Information contained herein supplements information of basic Flight Manual. For Limitations, Procedures, and Performance Data not contained in this supplement, or other applicable supplements, consult basic Flight Manual.

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Section 1

LIMITATIONS

1-6. WEIGHT AND CENTER OF GRAVITY

1-6-A. WEIGHT

Maximum approved internal gross weight for takeoff and landing is 5250 pounds (2381 kilograms) or as shown in IGE hover performance charts, Section 4.

CAUTION

LOADS THAT RESULT IN GW ABOVE 5,250 POUNDS (2381 KILOGRAMS) SHALL BE CARRIED ON THE CARGO HOOK.

1-6-B. CENTER OF GRAVITY

For longitudinal CG limits, refer to Gross Weight Longitudinal center of gravity limits charts (Figure 1-1).

For lateral CG limits, refer to Gross Weight Lateral center of gravity limits chart (Figure 1-2).

1-7. AIRSPEED

V_{NE} is 140 KIAS, sea level to 3000 feet H_D. Decrease V_{NE} for ambient conditions in accordance with AIRSPEED LIMITATIONS Placards and decals (Figure 1-3).

1-8. ALTITUDE

1-8-A. DENSITY

Maximum H_D for takeoff, landing, and in ground effect manuevers is 11,000 feet (3353 meters).

1-8-B. PRESSURE

Maximum operating pressure altitude is 20,000 feet (6096 meters).

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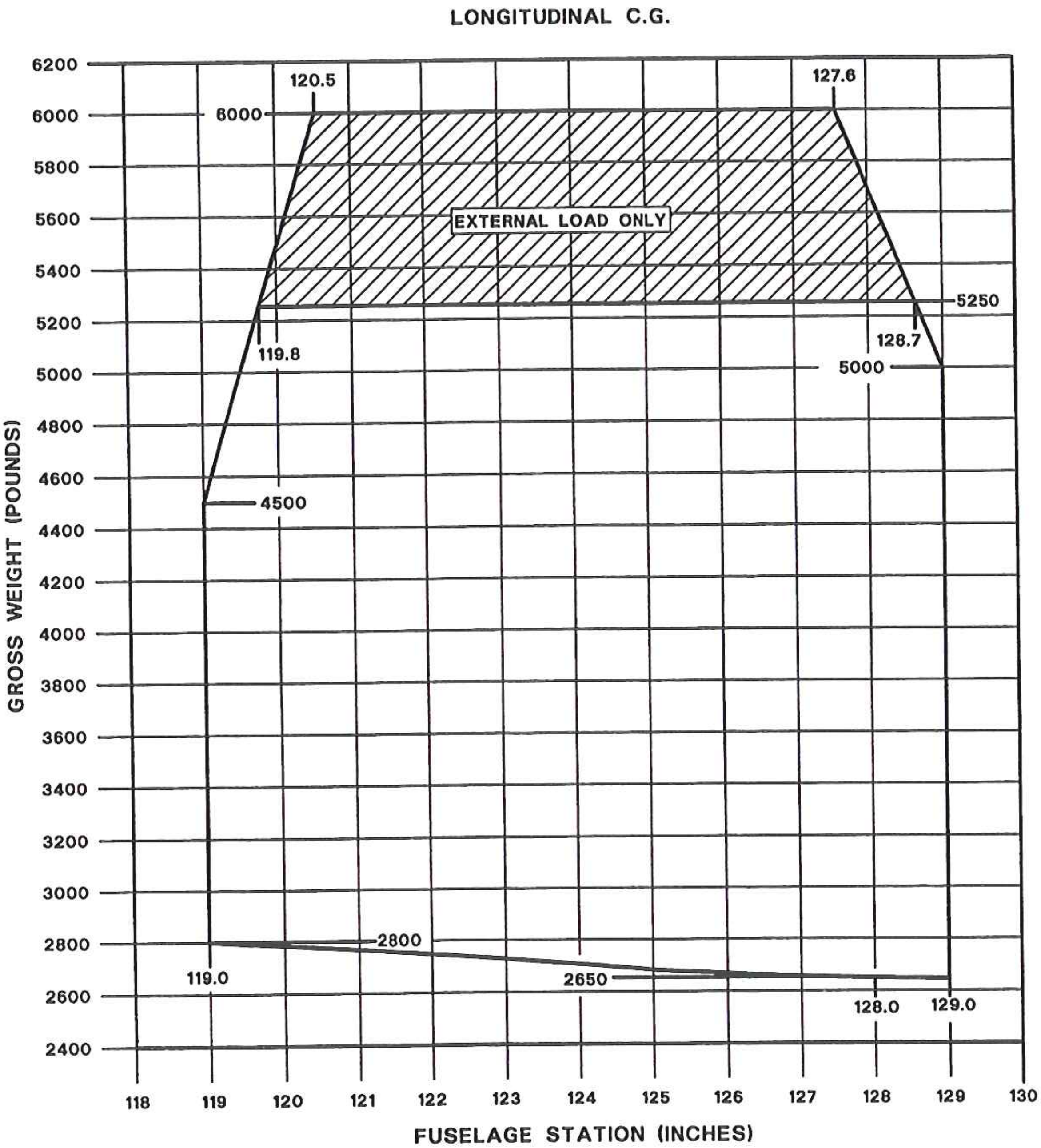

Figure 1-1. Gross Weight Longitudinal center of gravity limits (Sheet 1 of 2)

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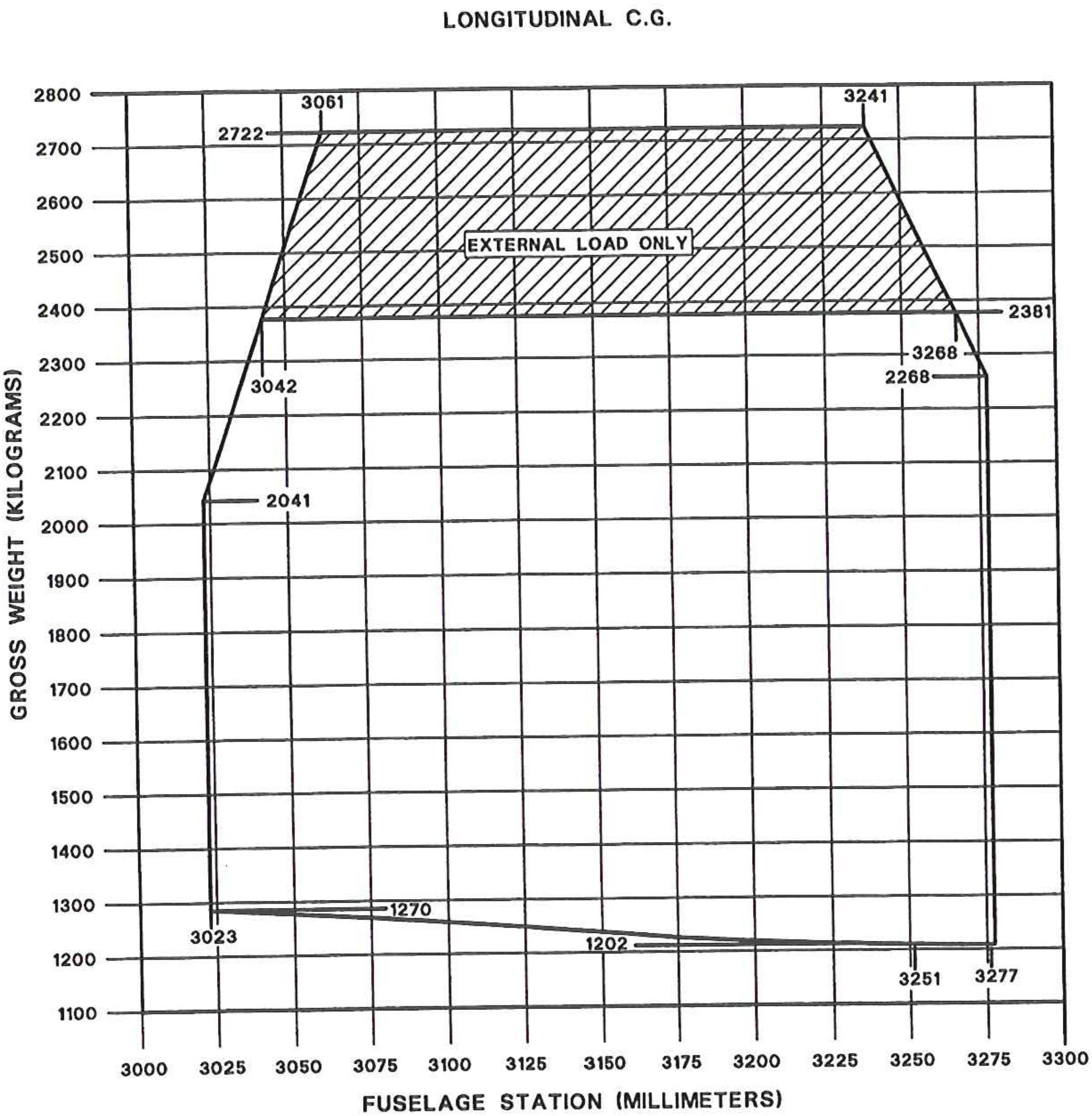

Figure 1-1. Gross Weight Longitudinal center of gravity limits (Sheet 2 of 2)

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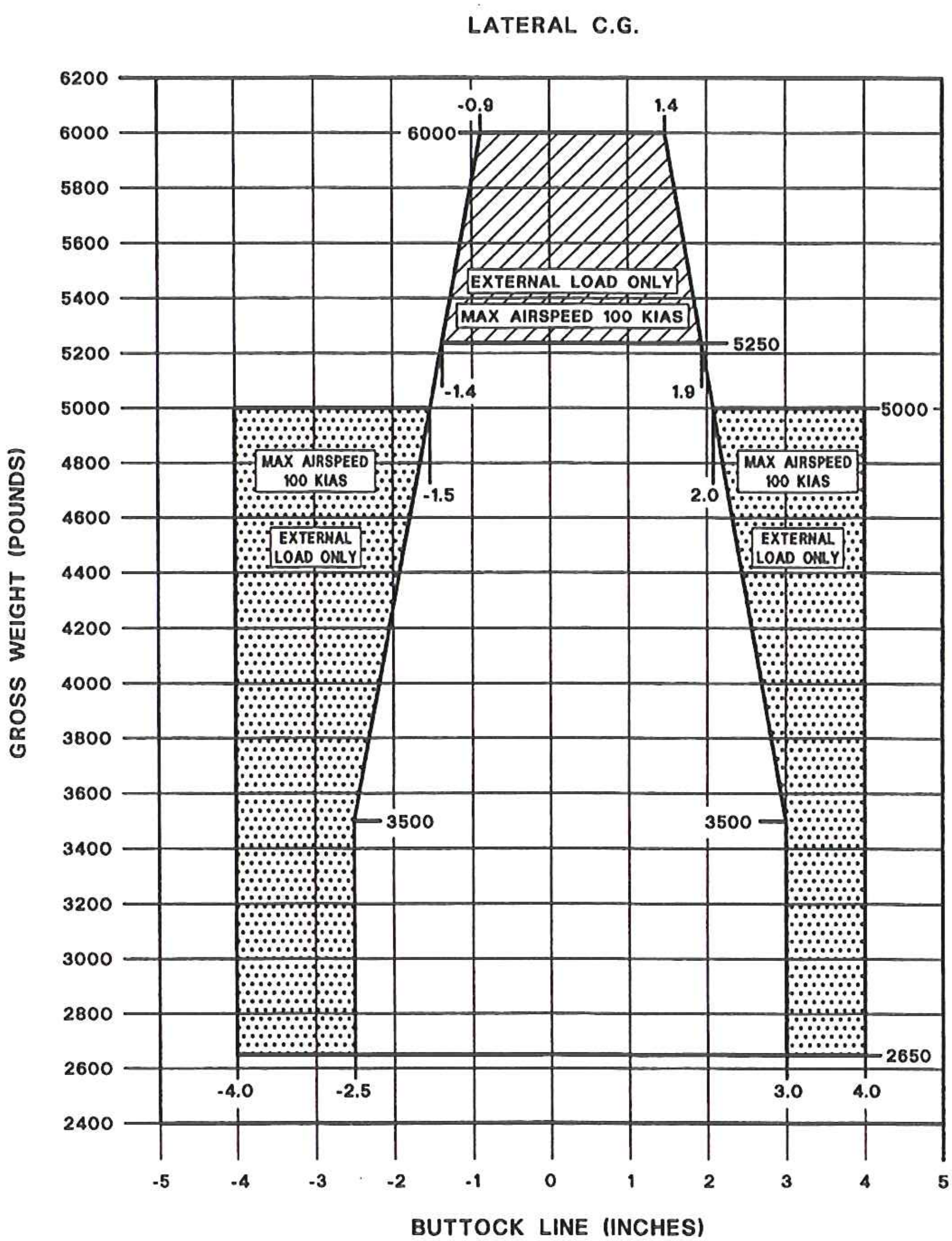

Figure 1-2. Gross Weight Lateral center of gravity limits (Sheet 1 of 2)

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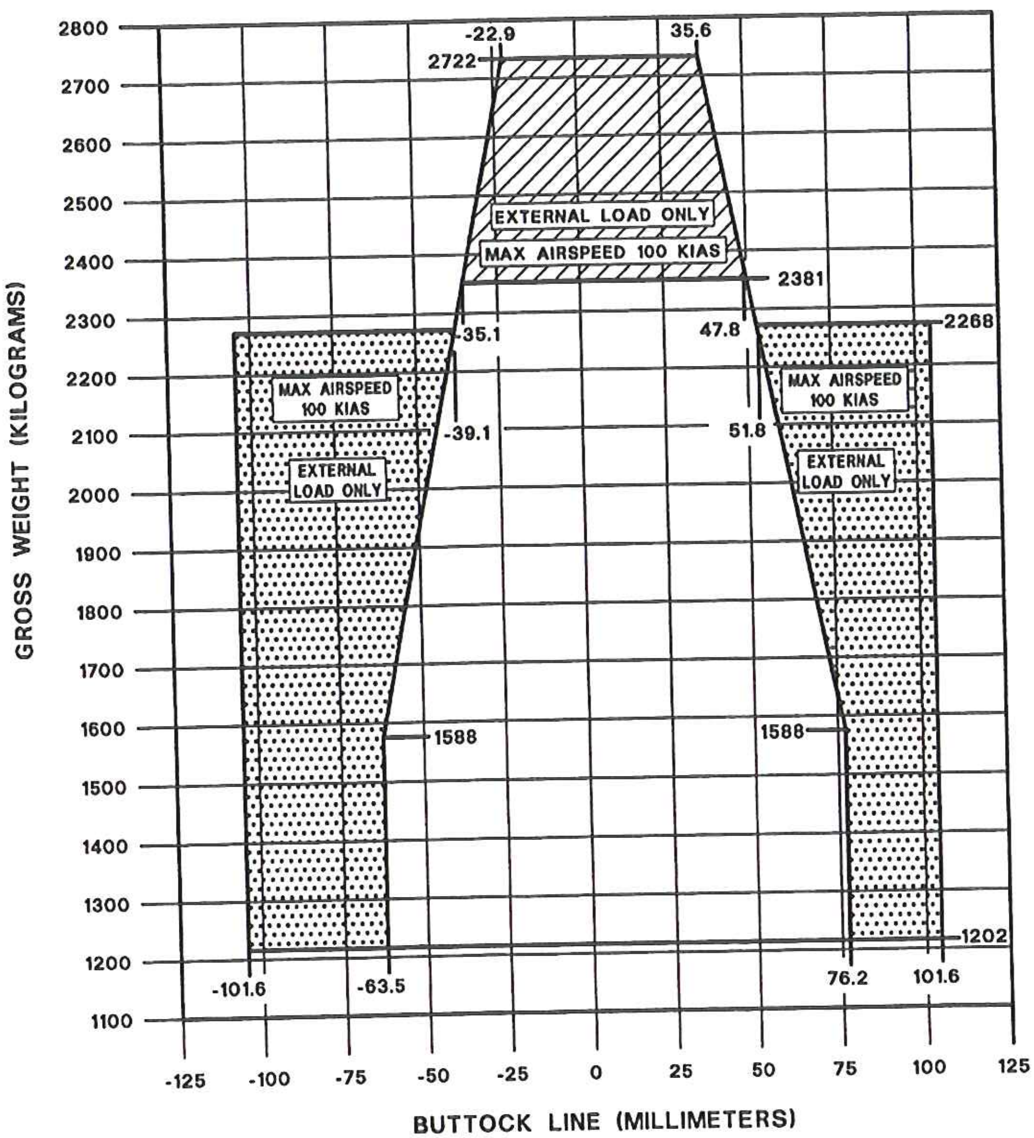
LATERAL C.G.



407FS28-1-2-2

Figure 1-2. Gross Weight Lateral center of gravity limits (Sheet 2 of 2)

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407 (5250 LB) AIRSPEED LIMITATIONS - KNOTS - IAS											
OAT C°	PRESSURE ALTITUDE FT X 1000										
	0	2	4	6	8	10	12	14	16	18	20
52	138	132	120	—	—	—	—	—	—	—	—
40	140	136	126	114	103	96	—	—	—	—	—
20	140	140	135	124	112	102	94	87	80	—	—
0	140	140	140	135	124	111	101	94	86	79	72
-20	140	140	140	139	134	123	111	101	93	86	78
-40	137	133	128	123	118	114	110	105	101	93	86
MAXIMUM AUTOROTATION VNE 100 KIAS											

407FS28-1-3

Figure 1-3. Placards and Decals (typical)

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Section 2

NORMAL PROCEDURES

No change from basic manual.

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Section 3

EMERGENCY/MALFUNCTION PROCEDURES

No change from basic manual.

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Section 4

PERFORMANCE

4-1. INTRODUCTION

Refer to appropriate performance charts in accordance with optional equipment installed.

4-4. HEIGHT - VELOCITY ENVELOPE

Altitude vs gross weight for height-velocity diagram (Figure 4-1) and Height-velocity (Figure 4-2) diagrams define conditions from which a safe landing can be made on a smooth, level, firm surface following an engine failure. Height velocity diagram is valid only when helicopter gross weight does not exceed limits of the Altitude vs Gross Weight diagram.

4-5. HOVER CEILING

NOTE

Hover performance charts are based on 100% ROTOR RPM.

Satisfactory stability and control have been demonstrated in each area of the Hover ceiling charts with winds as depicted on Hover ceiling wind accountability chart (refer to Basic Flight Manual).

Hover ceiling - in ground effect charts (Figures 4-3 and 4-4) and Hover ceiling - out of ground effect charts (Figures 4-5 and 4-6) present hover performance as allowable gross weight for conditions of H_P and OAT. These hovering weights are obtainable in zero wind conditions. Each chart is divided into two areas. Area A (non shaded area) of hover ceiling charts presents hover performance (relative to GW) for conditions where adequate control margins exist for all relative wind conditions up to 35 knots, for hover, takeoff, and landing. Area B (shaded area) of hover ceiling charts presents hover performance (relative to GW) where adequate control margins exist for relative winds within ± 45° of nose of helicopter up to 35 knots, for lateral CG not exceeding ±2.5 inches (±63 mm); and up to 17 knots, for lateral CG to ±4.0 inches (±102 mm); for hover, takeoff, and landing.

4-7. CLIMB AND DESCENT

4-7-A. RATE OF CLIMB

Rate of climb (takeoff power) charts are presented in Figure 4-7, and Rate of climb (maximum continuous power) charts are presented in Figure 4-8.

4-10. NOISE LEVELS

4-10-A. FAR PART 36 STAGE 2 NOISE LEVEL

Model 407 is certified as a Stage 2 helicopter as prescribed in FAR Part 36, Subpart H, for gross weights up to and including certificated maximum takeoff and landing weight of 5250 pounds (2382 kilograms).

Certified flyover noise level for Model 407 is 85.5 dBA SEL.

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4-10-B. CANADIAN AIRWORTHINESS MANUAL CHAPTER 516 AND ICAO ANNEX 16 NOISE LEVEL

Model 407 complies with noise emission standards applicable to helicopter as set out by International Civil Aviation Organization (ICAO) in Annex 16, Volume 1, Chapter 11, for gross weights up to and including certified maximum takeoff and landing weight of 5250 pounds (2382 kilograms).

Flyover noise level for Model 407 is 85.5 dBA SEL.

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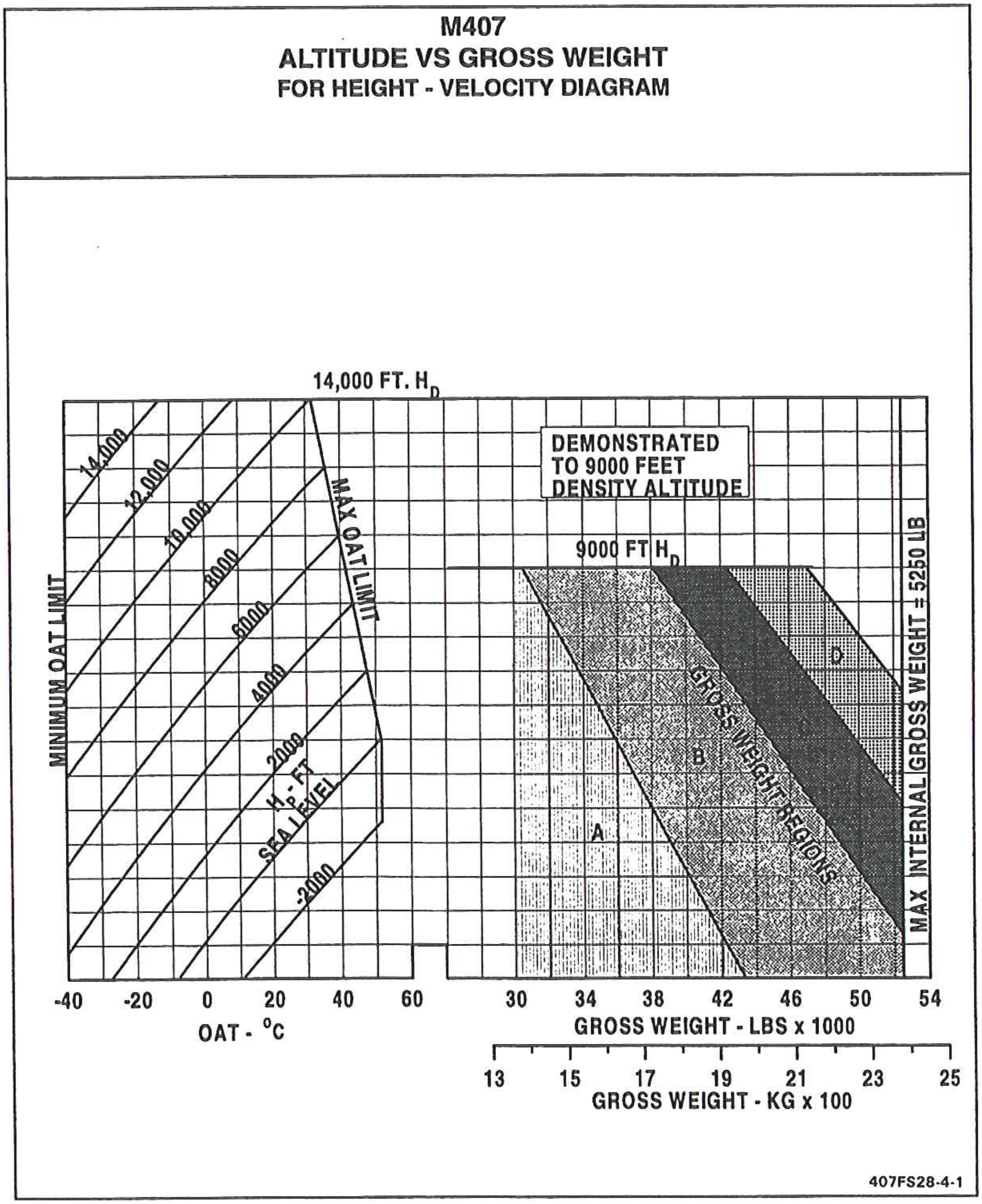

Figure 4-1. Altitude vs gross weight for height - velocity diagram

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HEIGHT - VELOCITY DIAGRAM FOR GROSS WEIGHT REGIONS A TO D

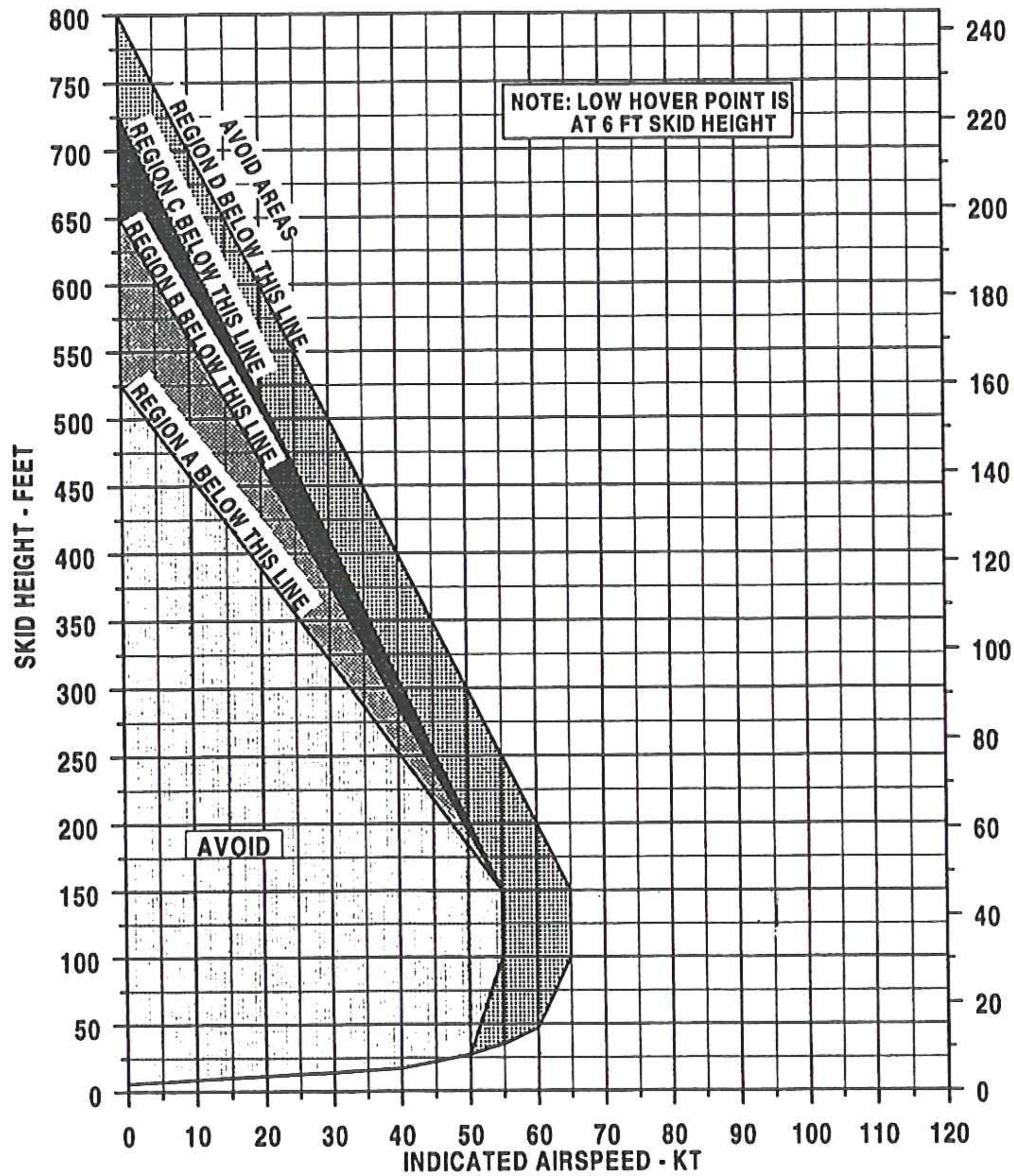

Figure 4-2. Height - velocty diagram

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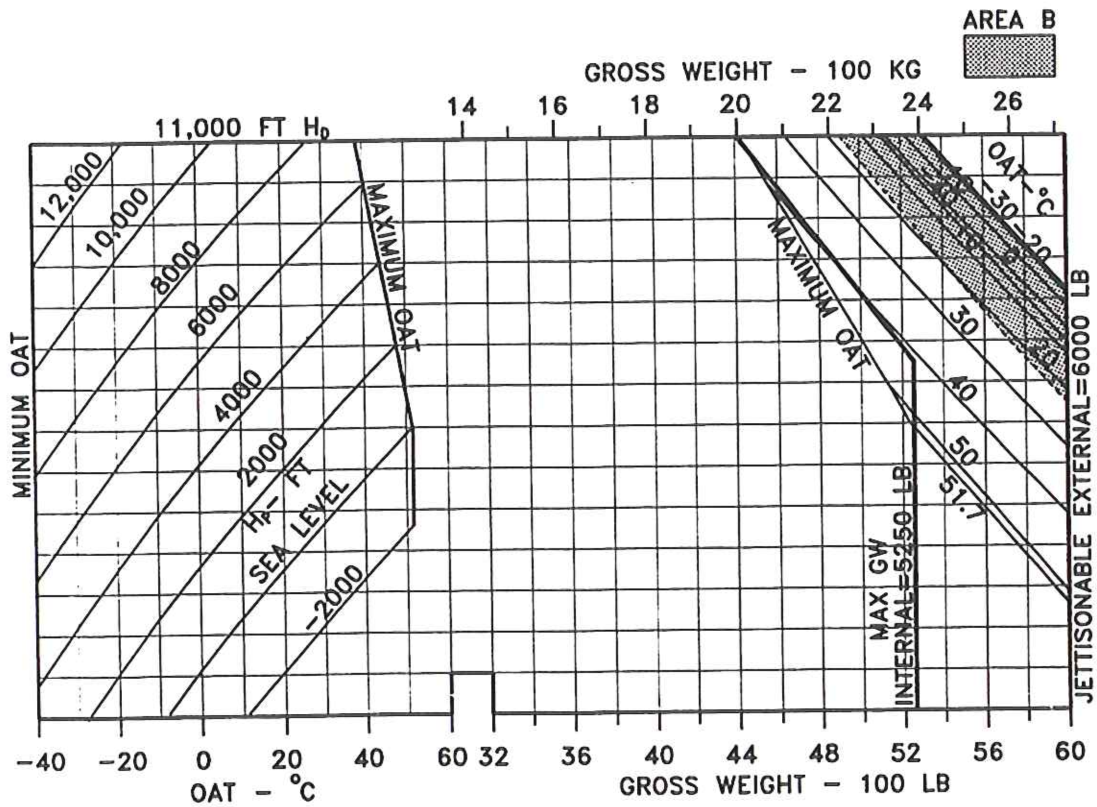
HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET



407FS28-4-3-1

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 1 of 16)

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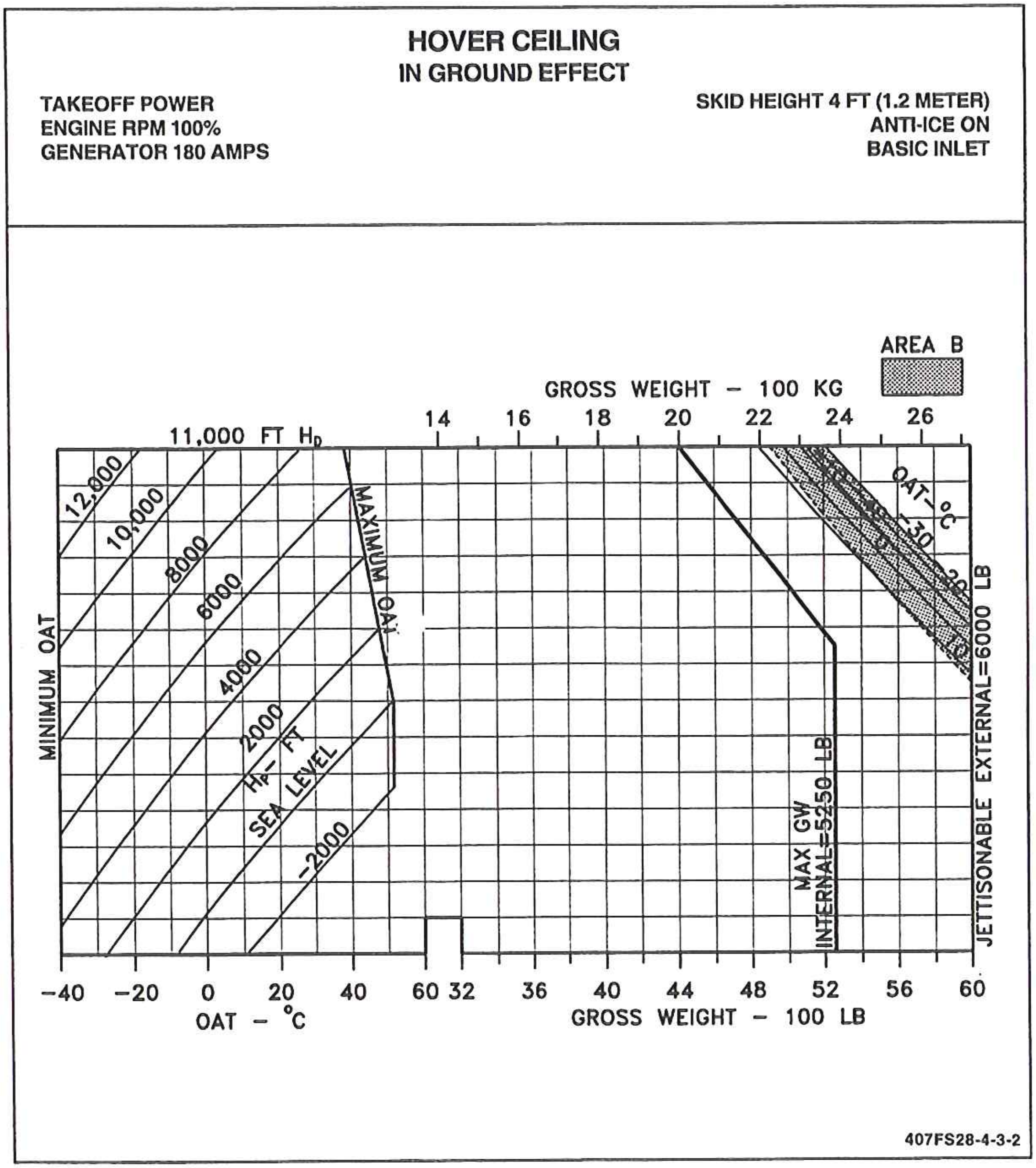

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 2 of 16)

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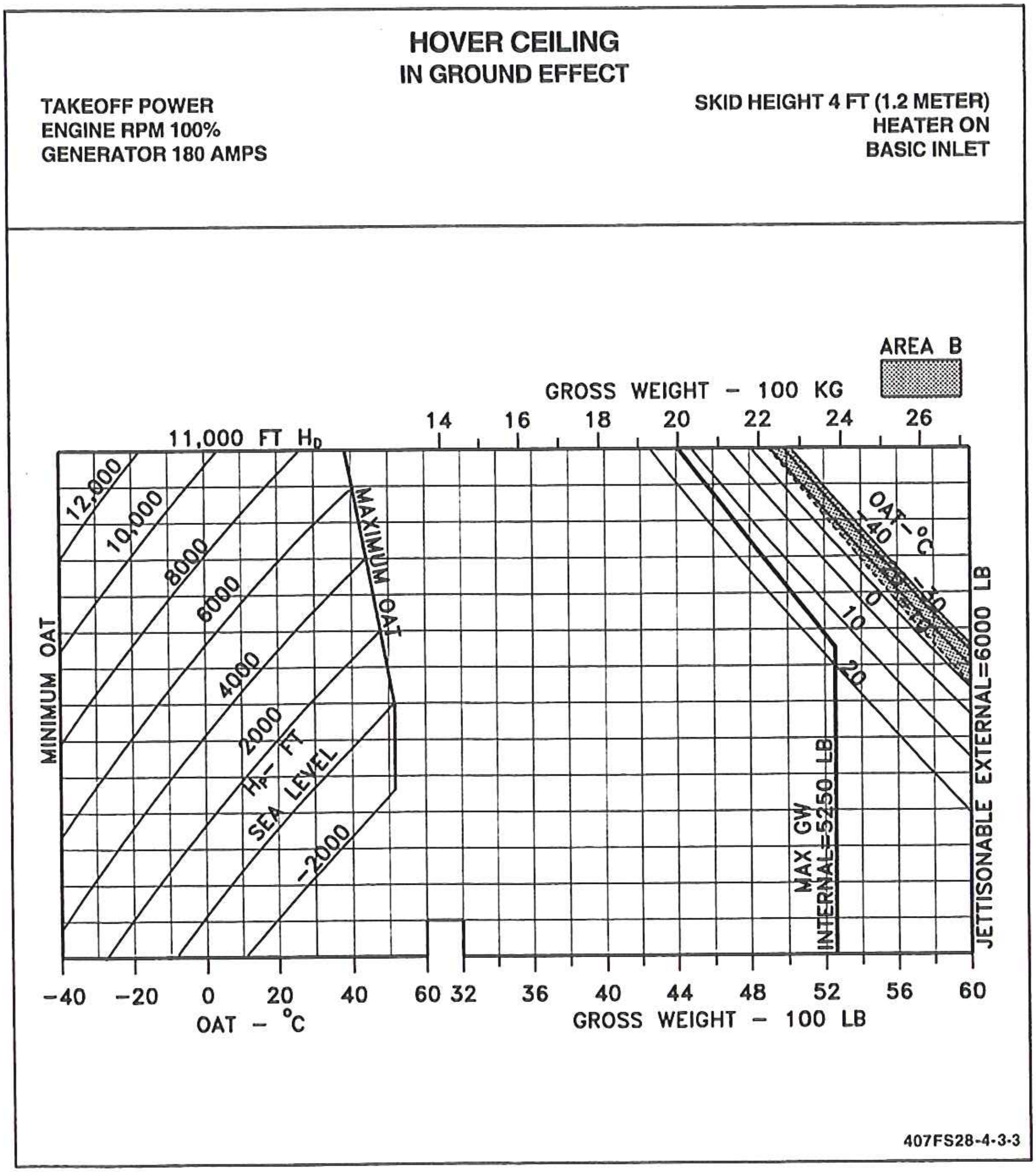

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 3 of 16)

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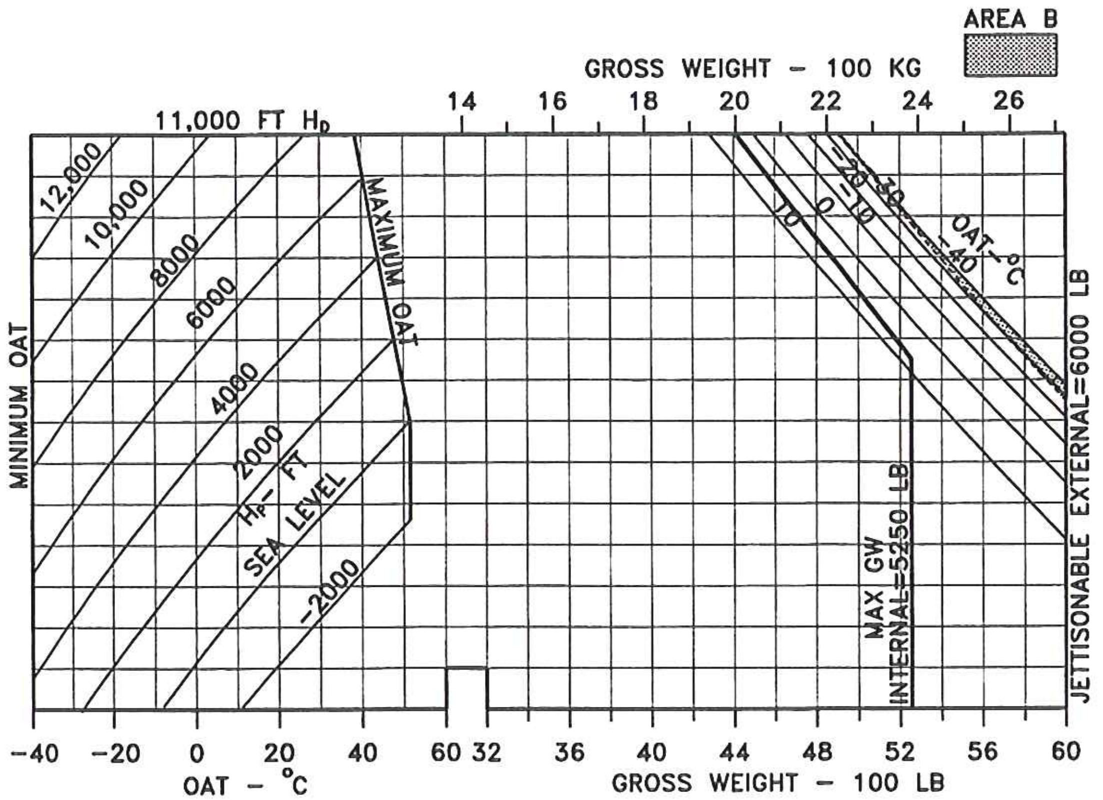
HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET



407FS28-4-3-4

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 4 of 16)

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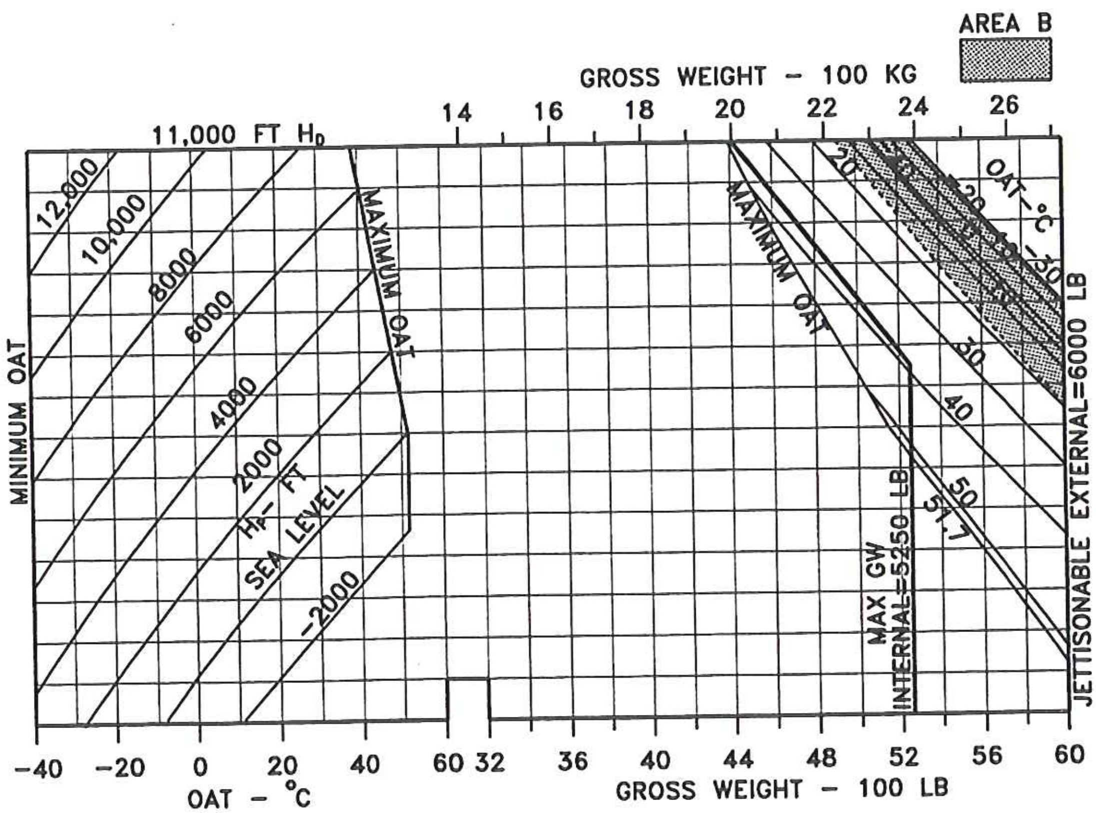
HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR



407FS28-4-3-5

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 5 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
PARTICLE SEPARATOR

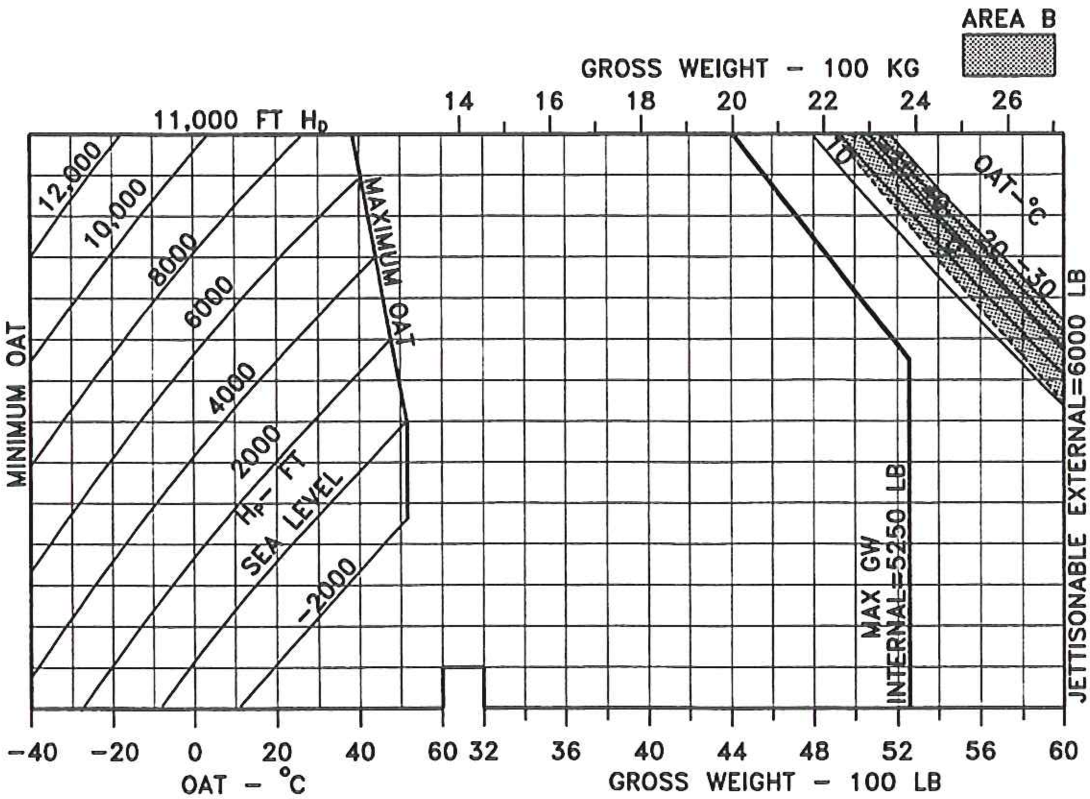

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Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 6 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
BASIC INLET
SNOW BAFFLES

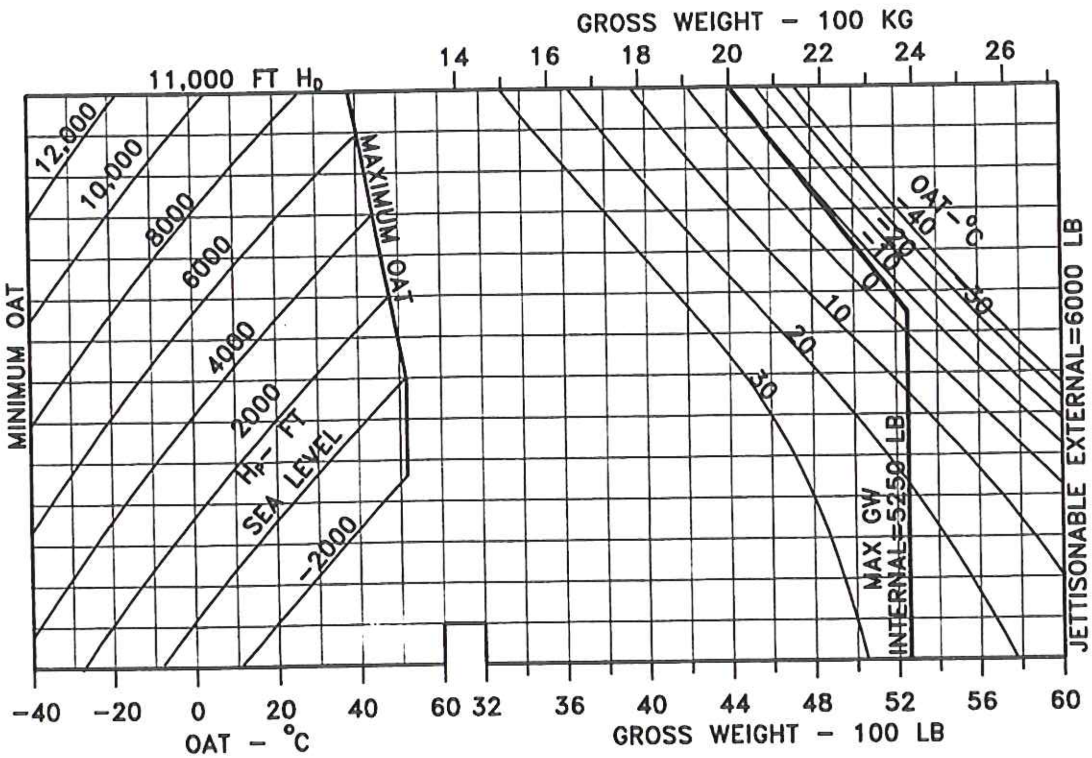

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 10 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
PARTICLE SEPARATOR

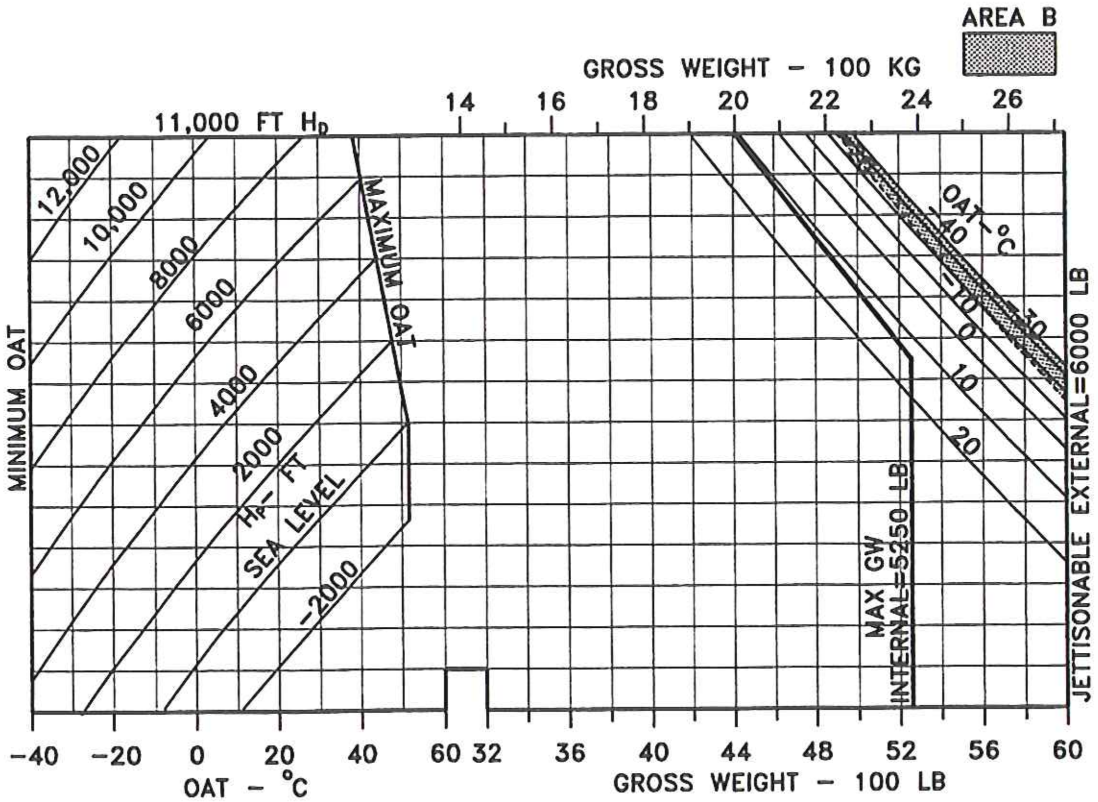

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Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 7 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR

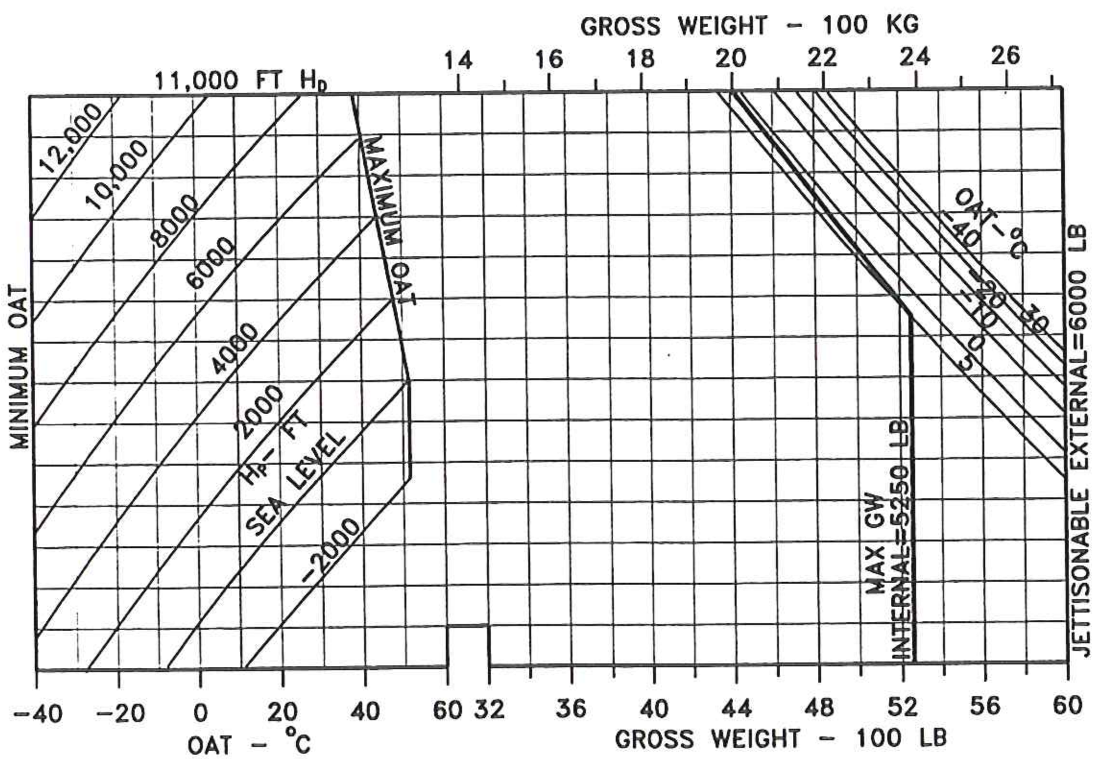

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Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 8 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET
SNOW BAFFLES

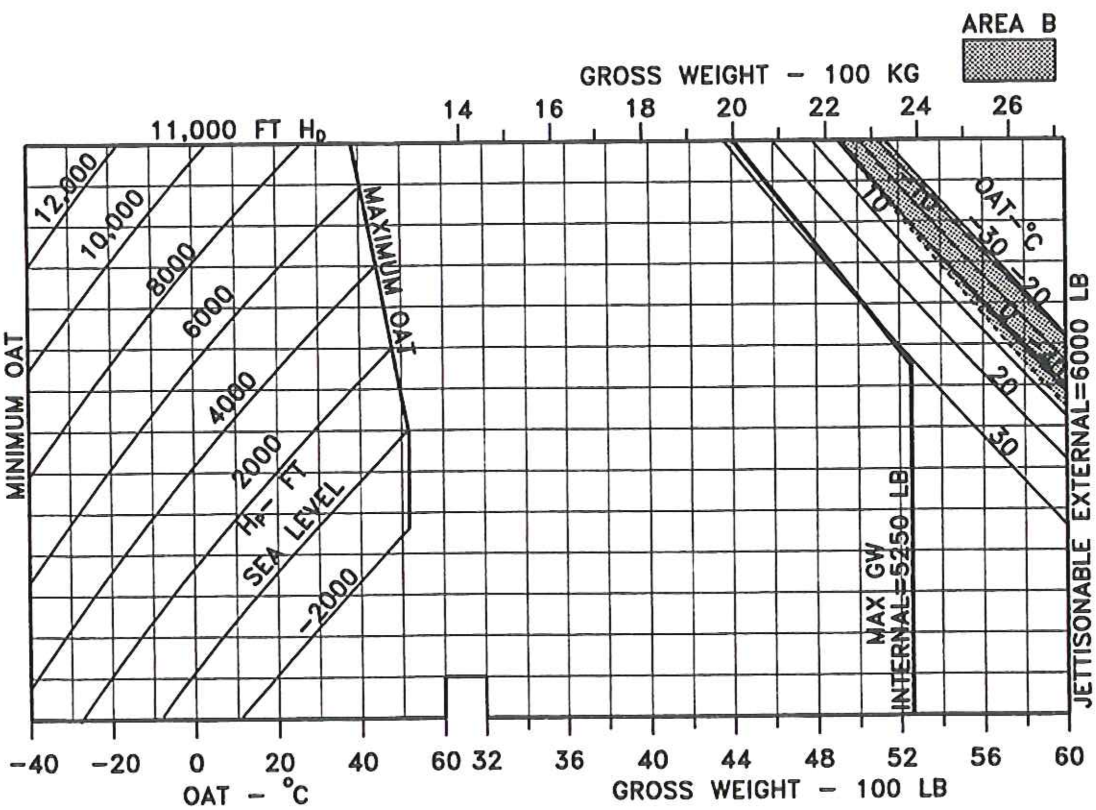

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 9 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
BASIC INLET
SNOW BAFFLES

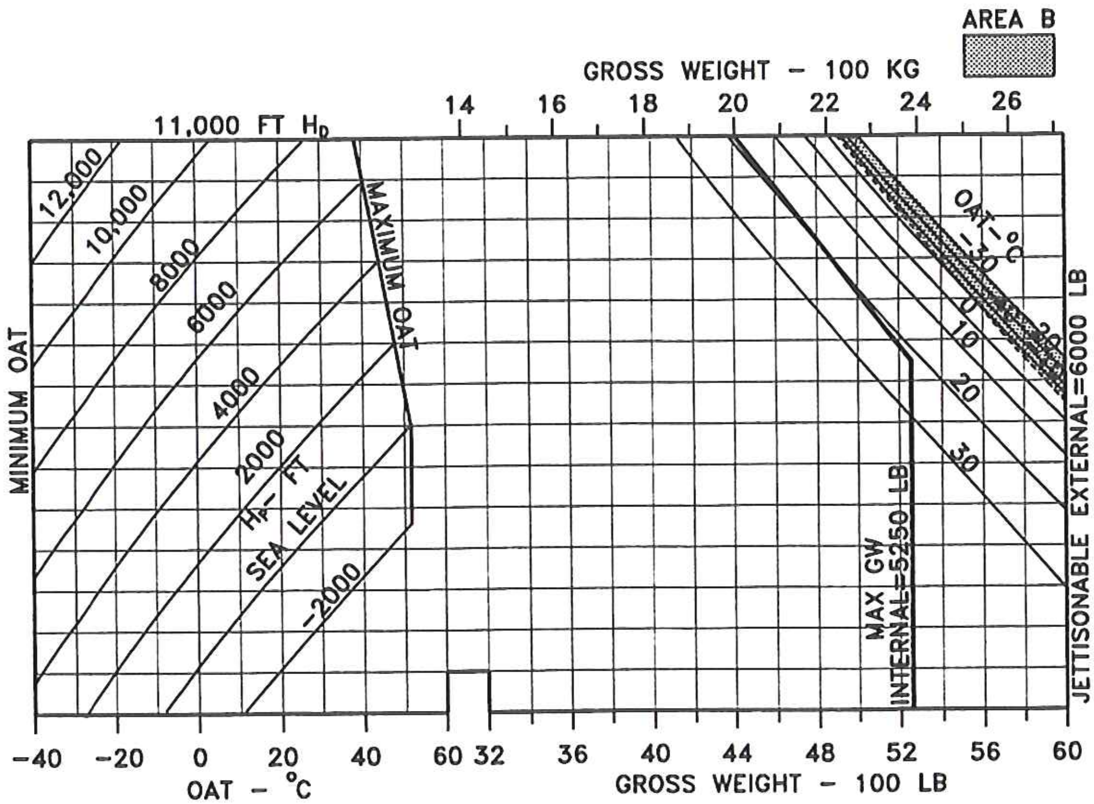

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 10 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
BASIC INLET
SNOW BAFFLES

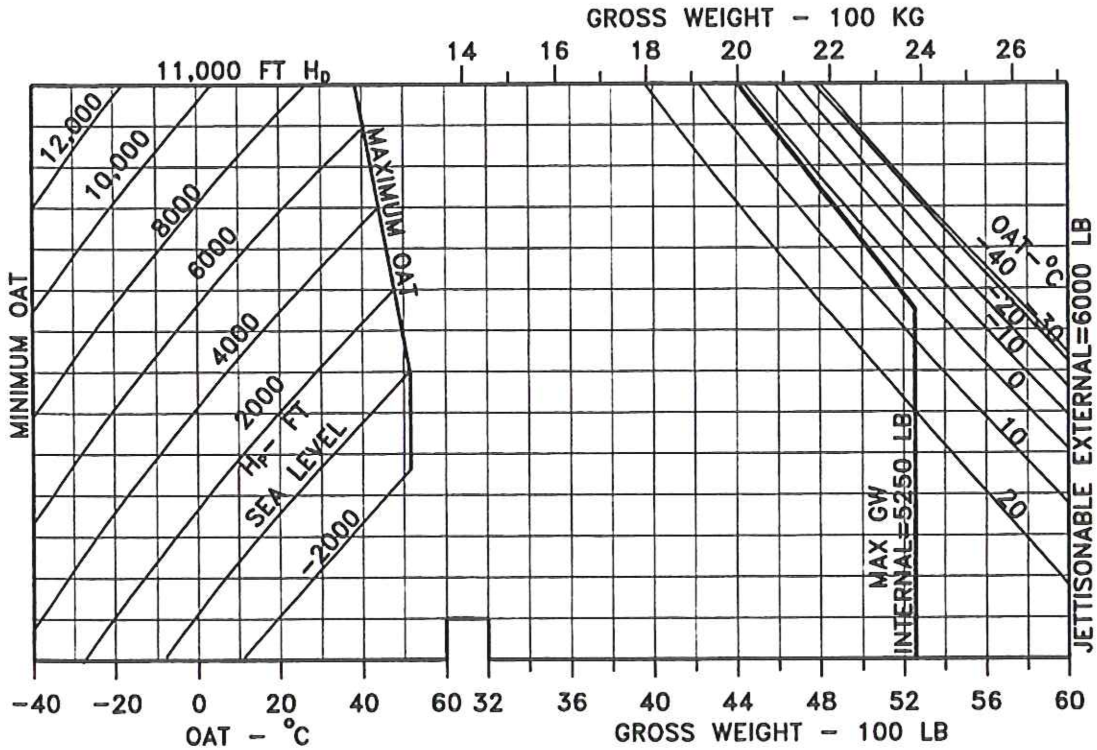

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Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 11 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET
SNOW BAFFLES

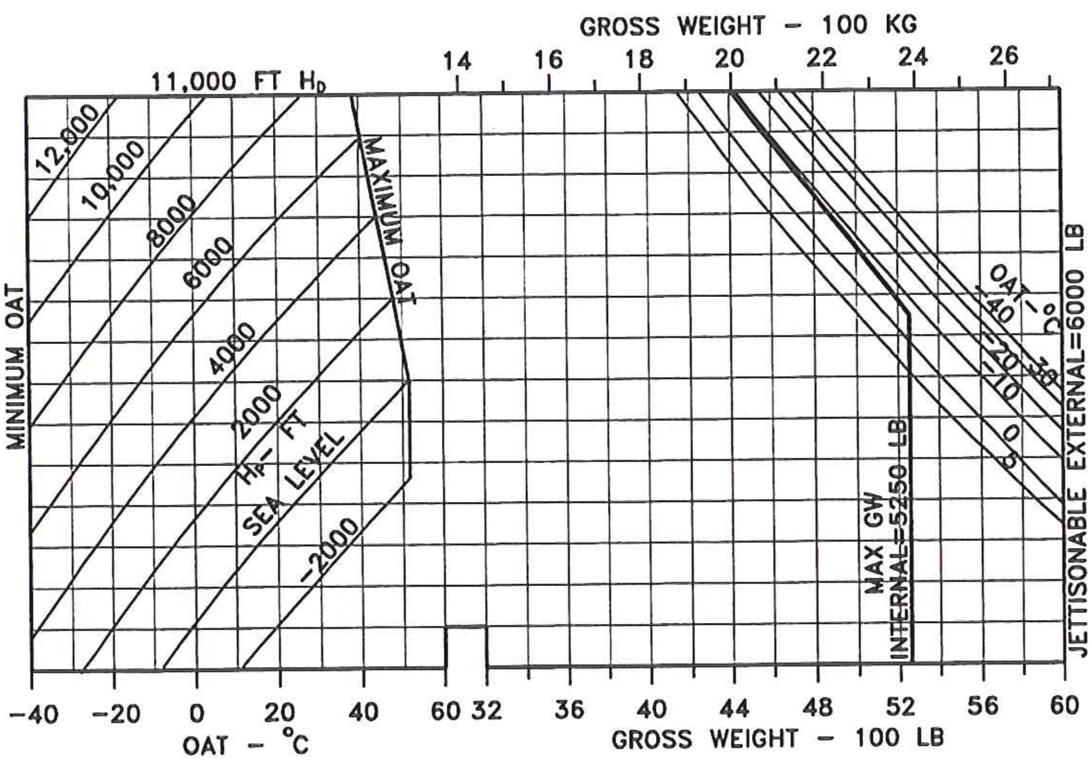

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Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 12 of 16)

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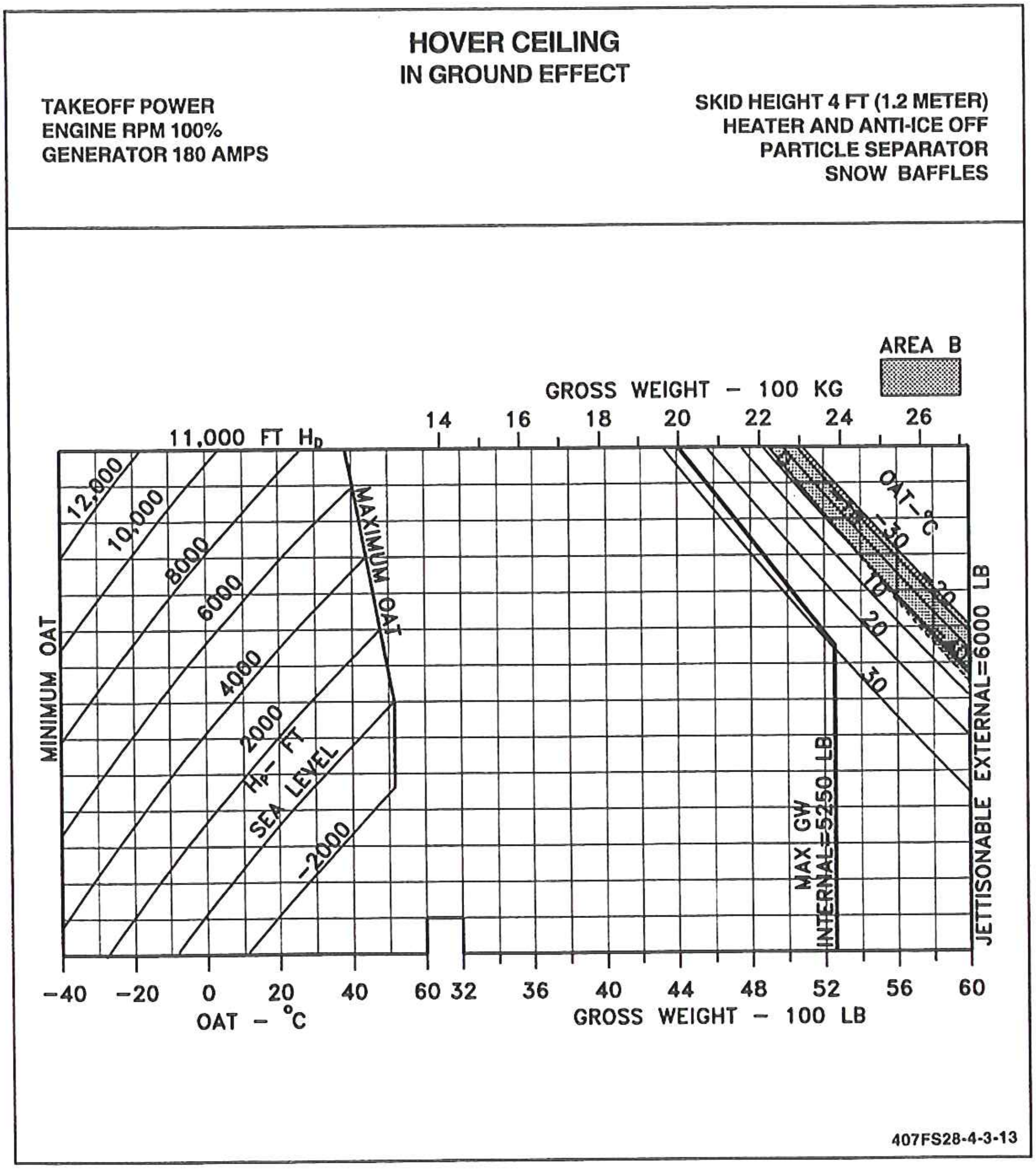

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 13 of 16)

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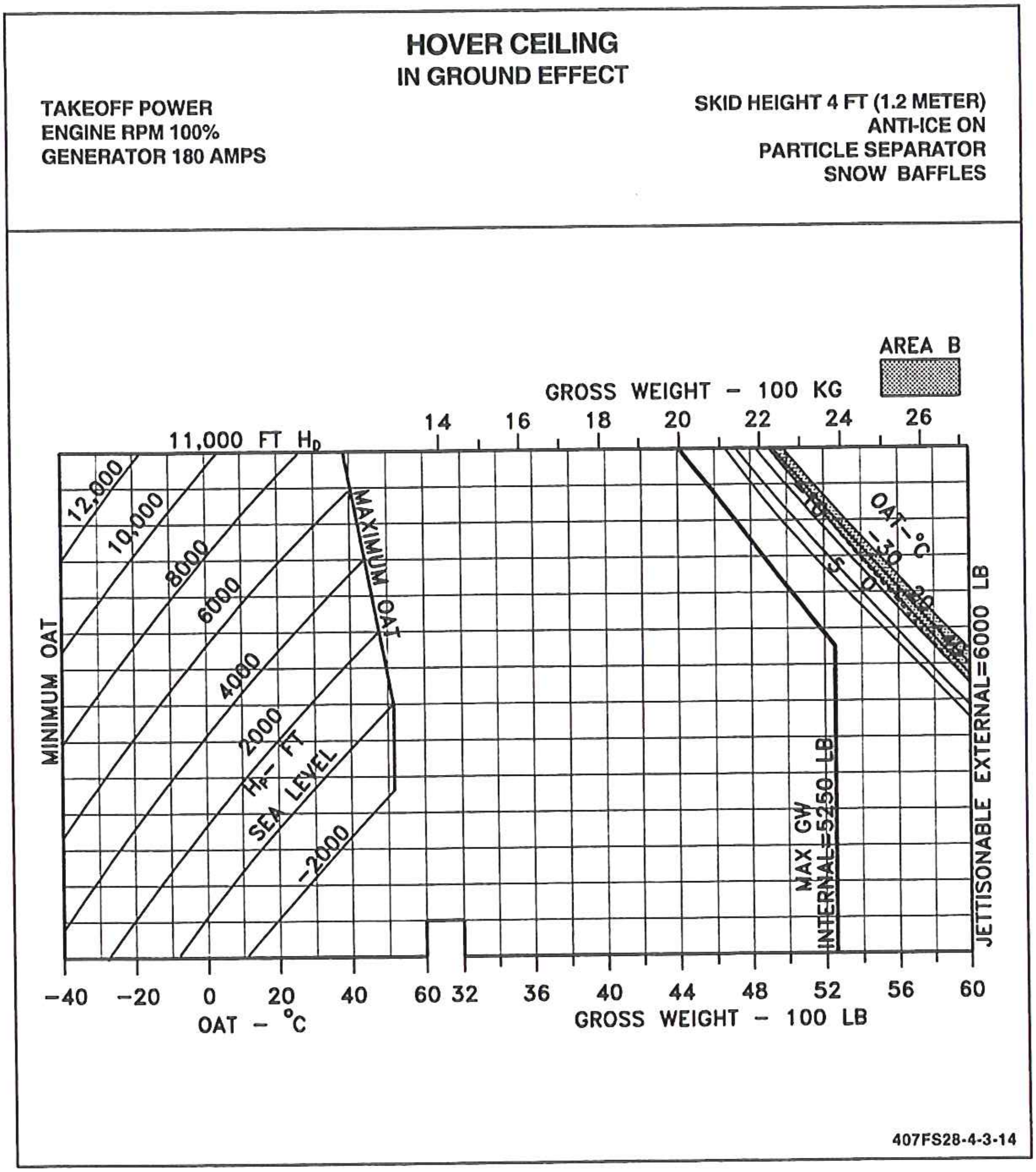

Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 14 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
PARTICLE SEPARATOR
SNOW BAFFLES

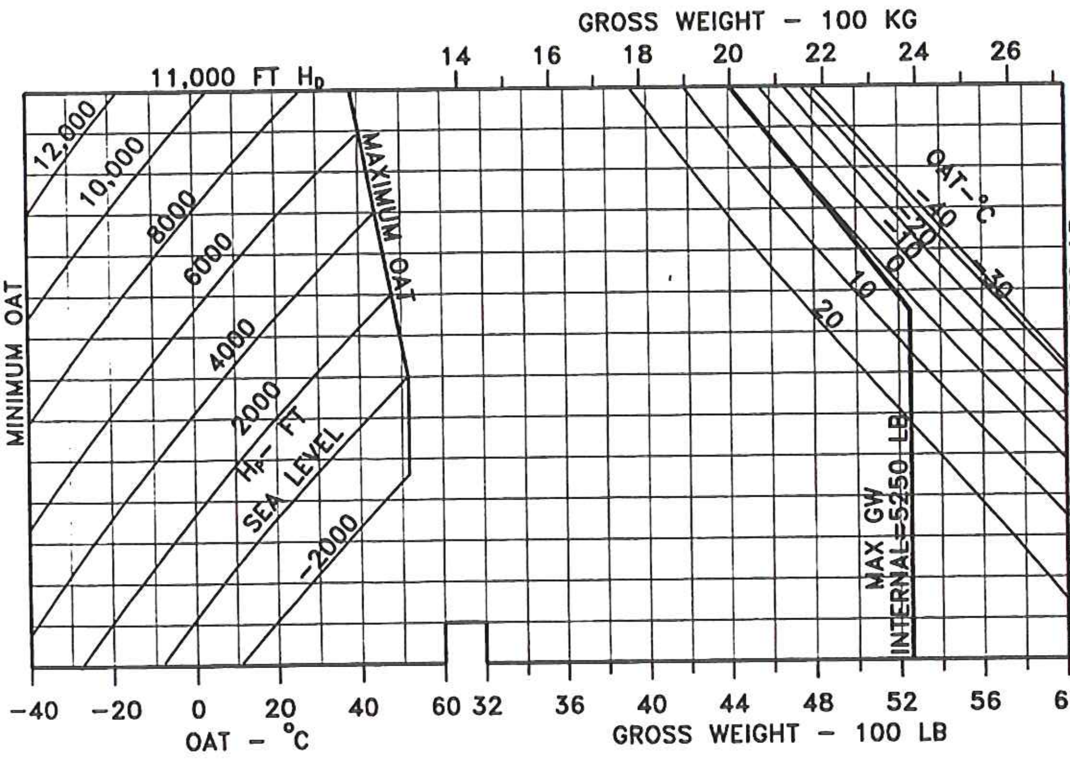

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Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 15 of 16)

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HOVER CEILING
IN GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR
SNOW BAFFLES

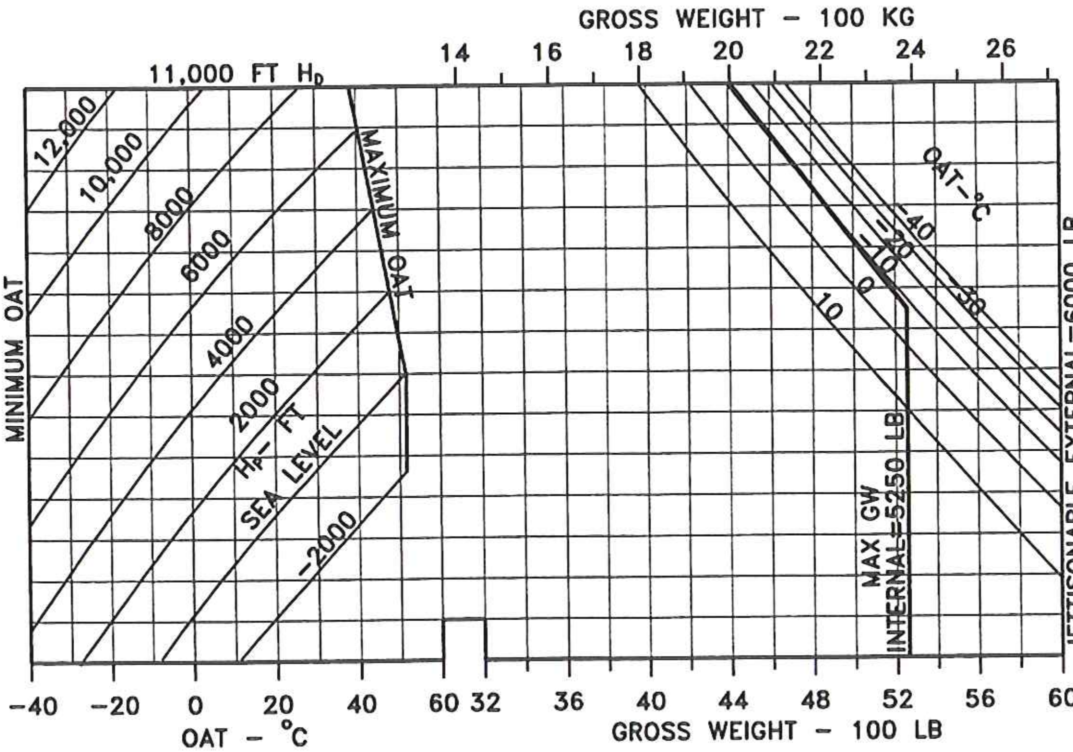

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Figure 4-3. Hover ceiling IGE - takeoff power (Sheet 16 of 16)

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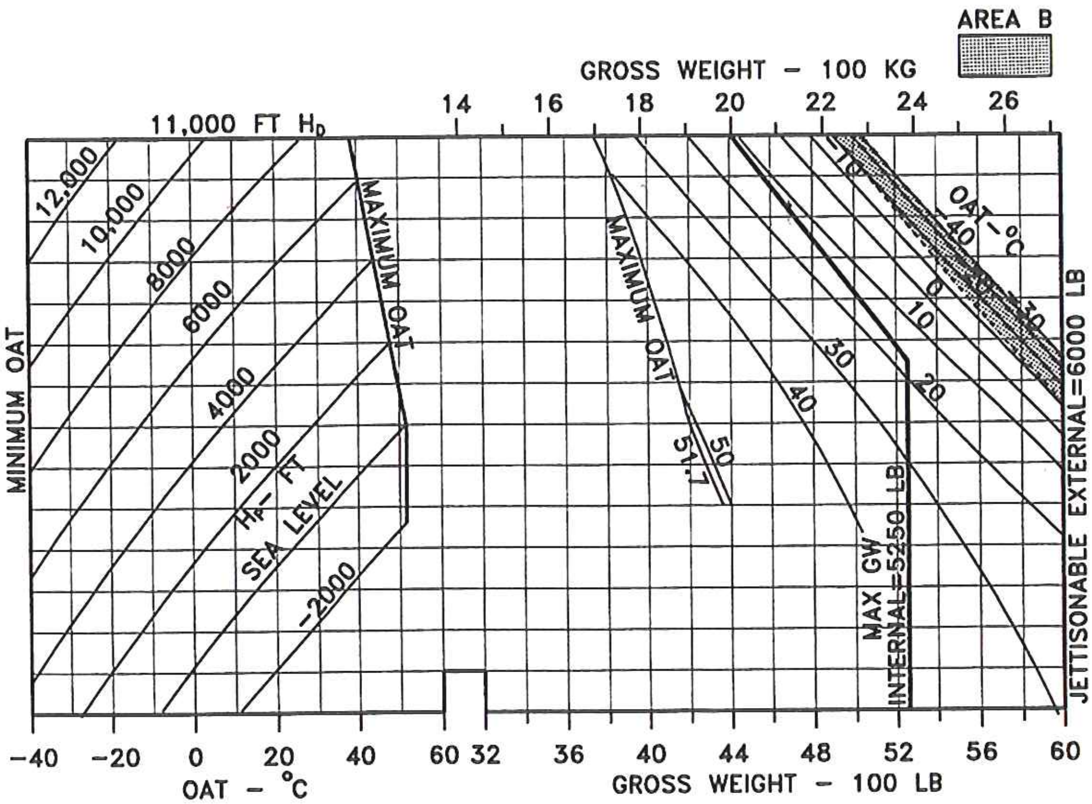
HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET



407FS28-4-4-1

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 1 of 16)

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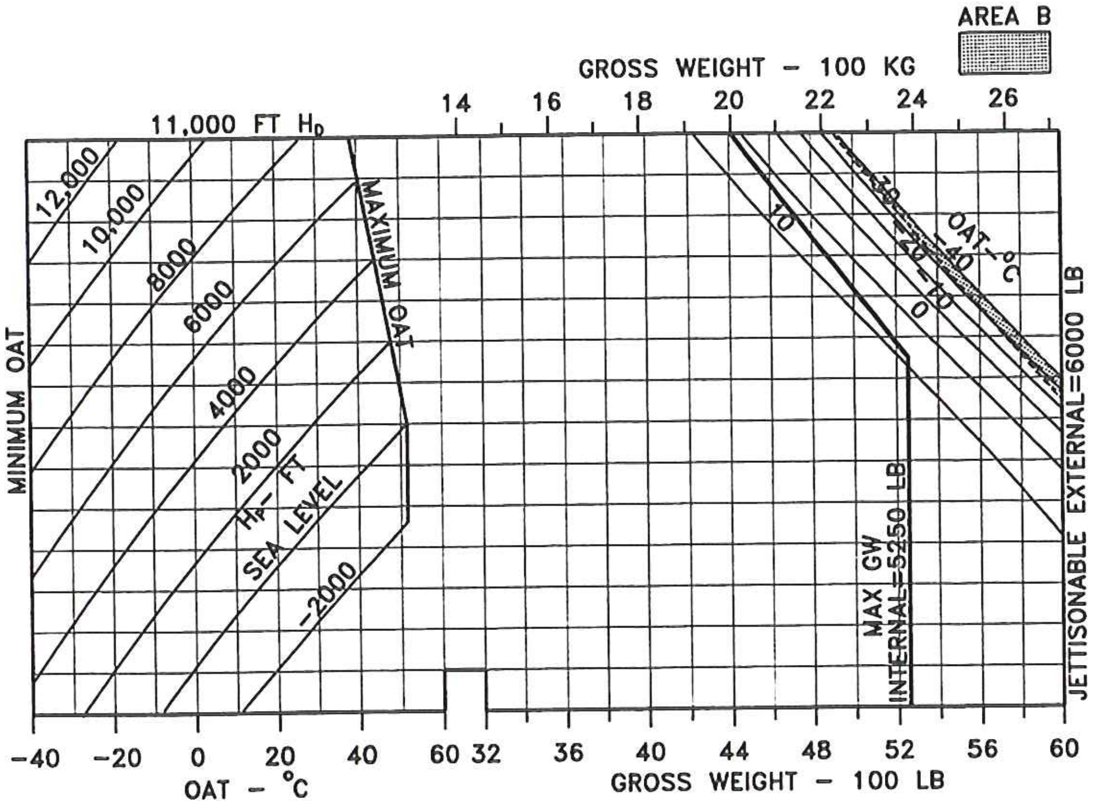
HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
BASIC INLET



407FS28-4-4-2

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 2 of 16)

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HOVER CEILING

IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
BASIC INLET

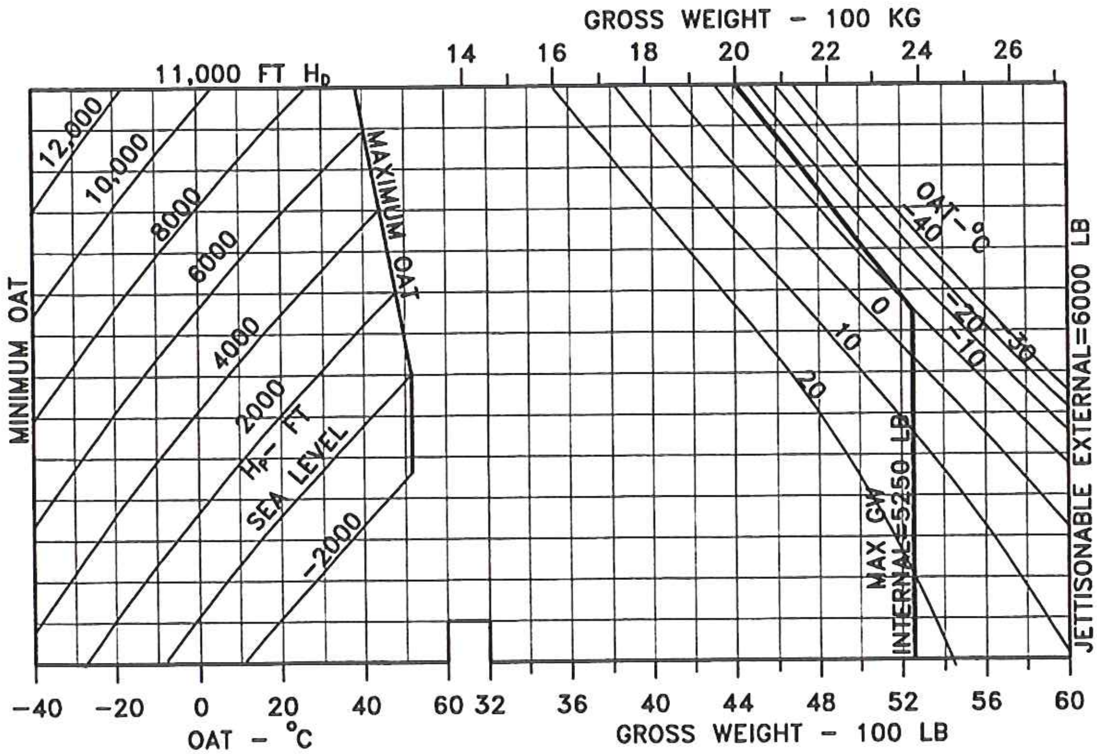

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 3 of 16)

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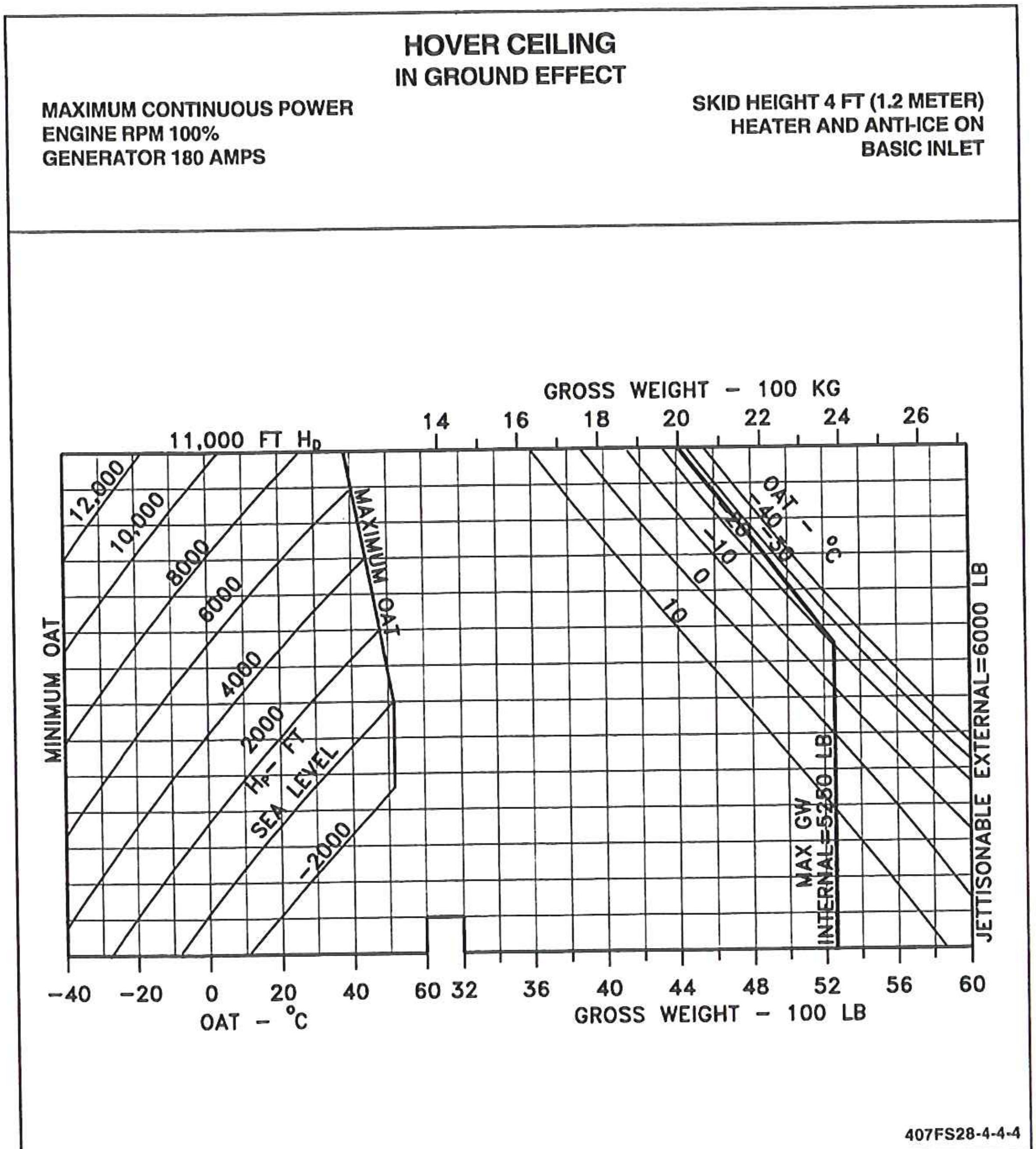

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 4 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR

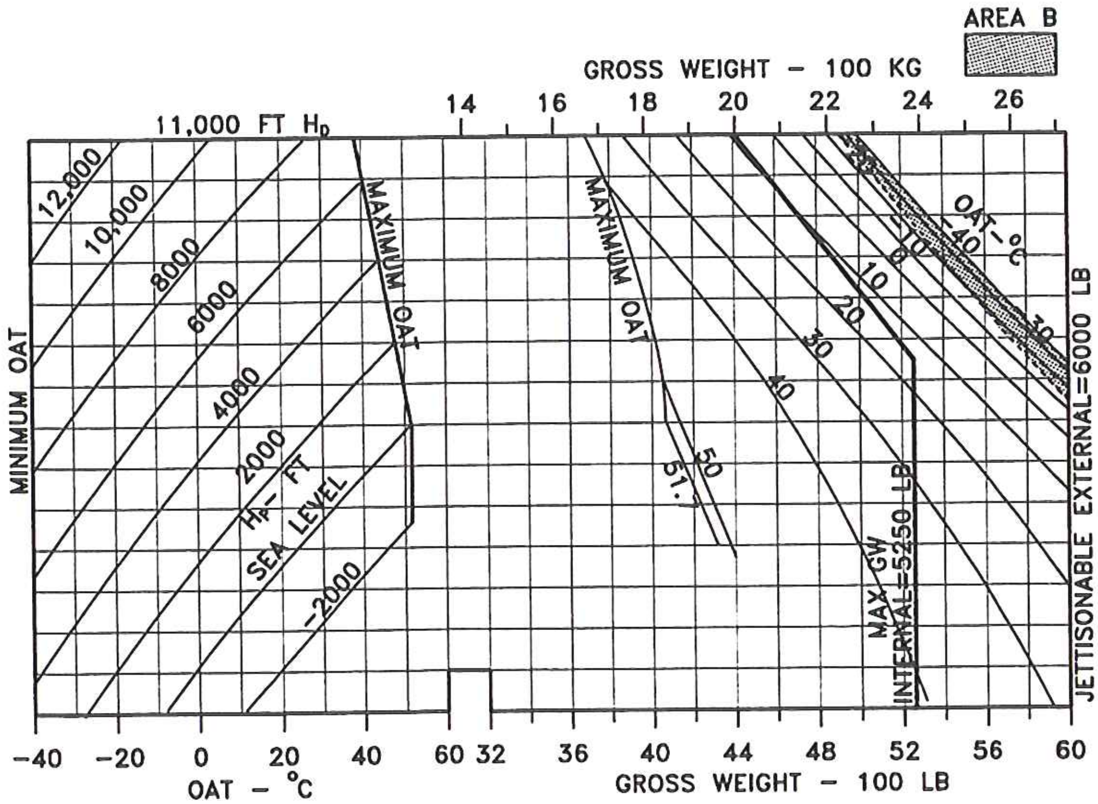

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 5 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
PARTICLE SEPARATOR

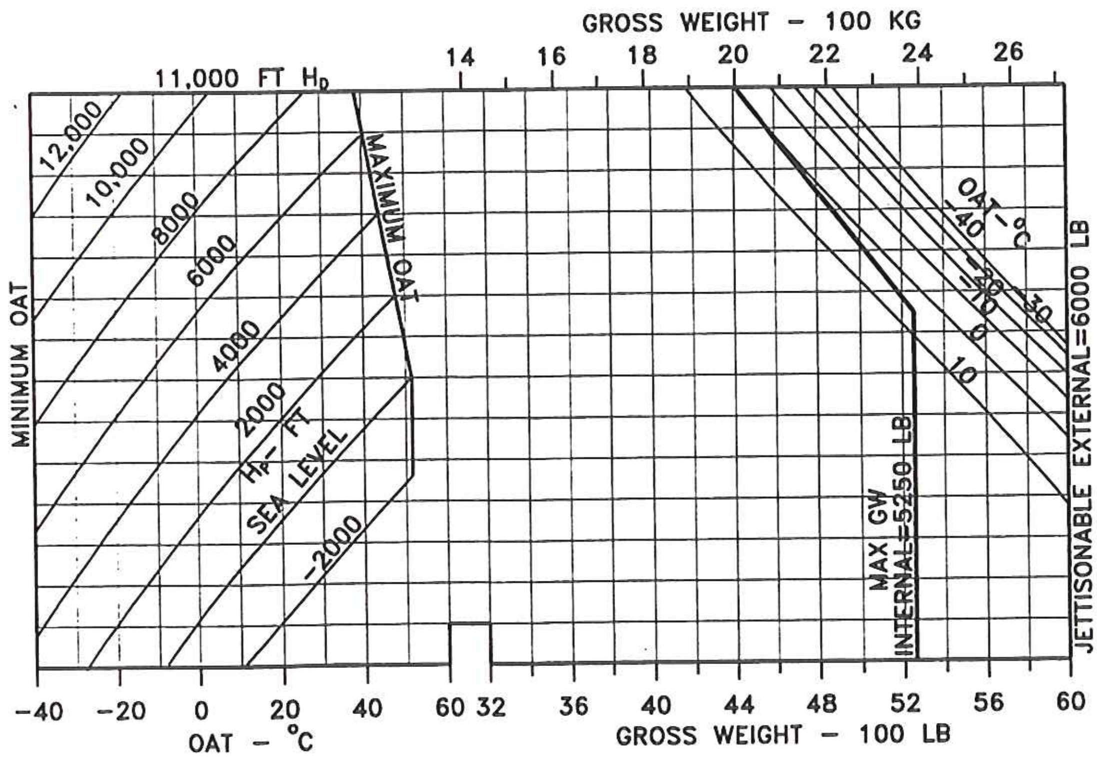

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 6 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
PARTICLE SEPARATOR

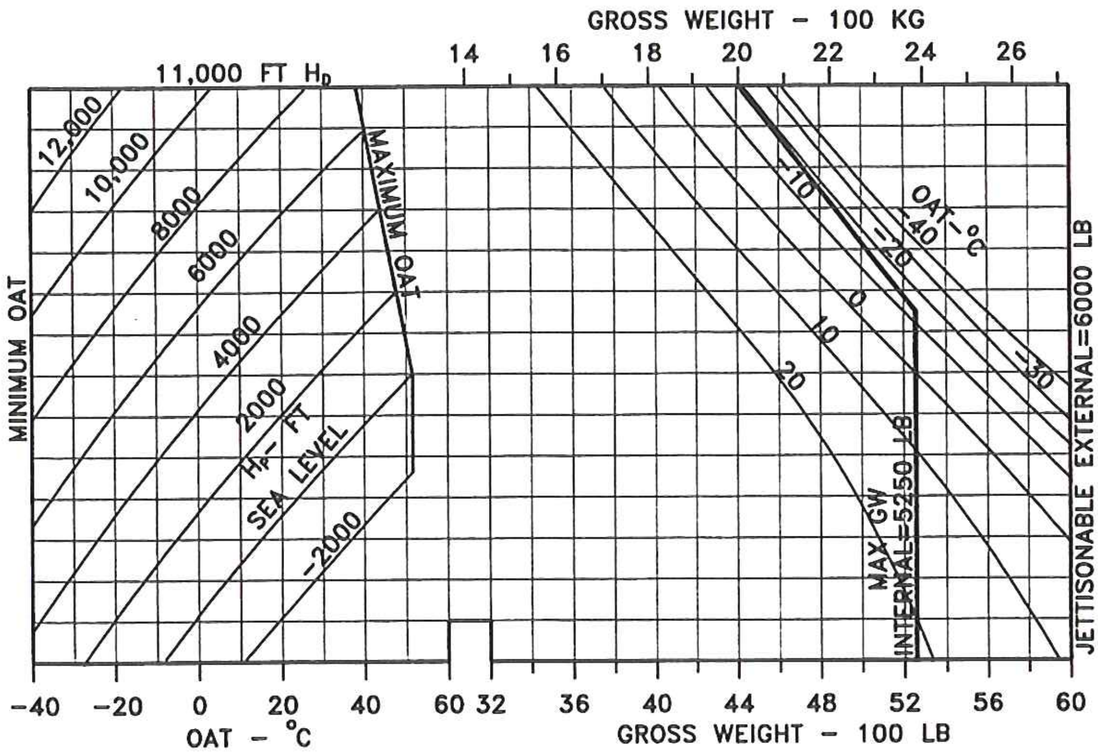

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 7 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR

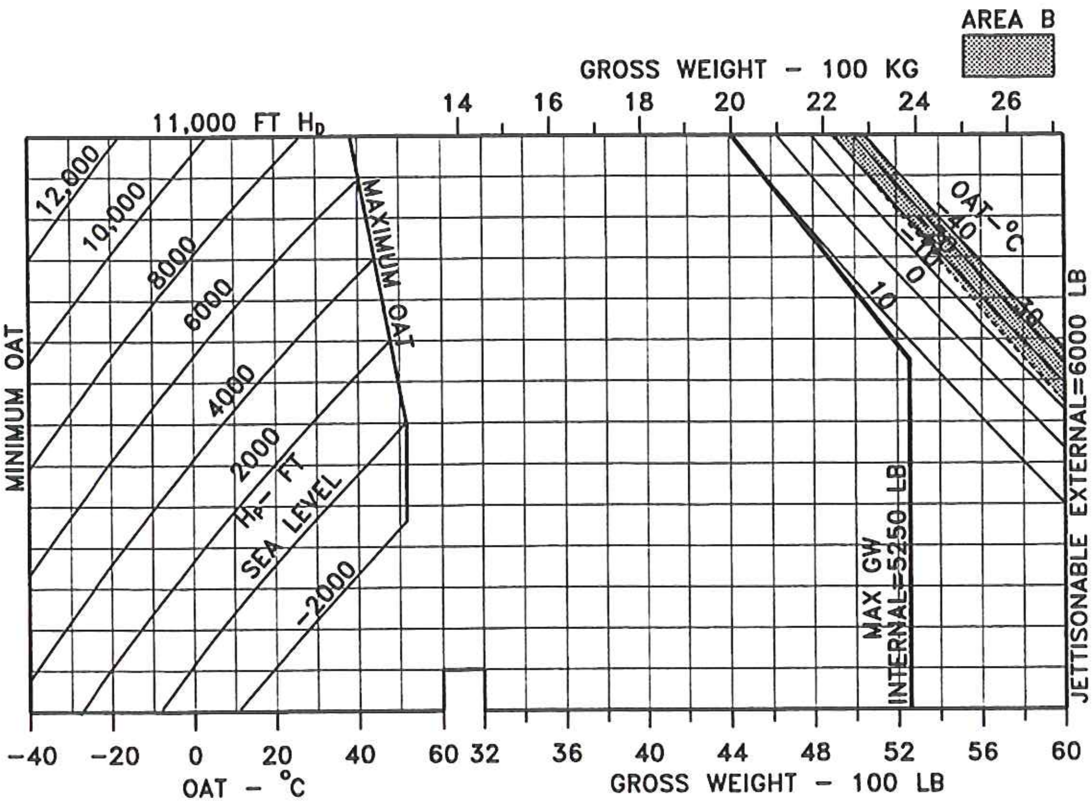

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 8 of 16)

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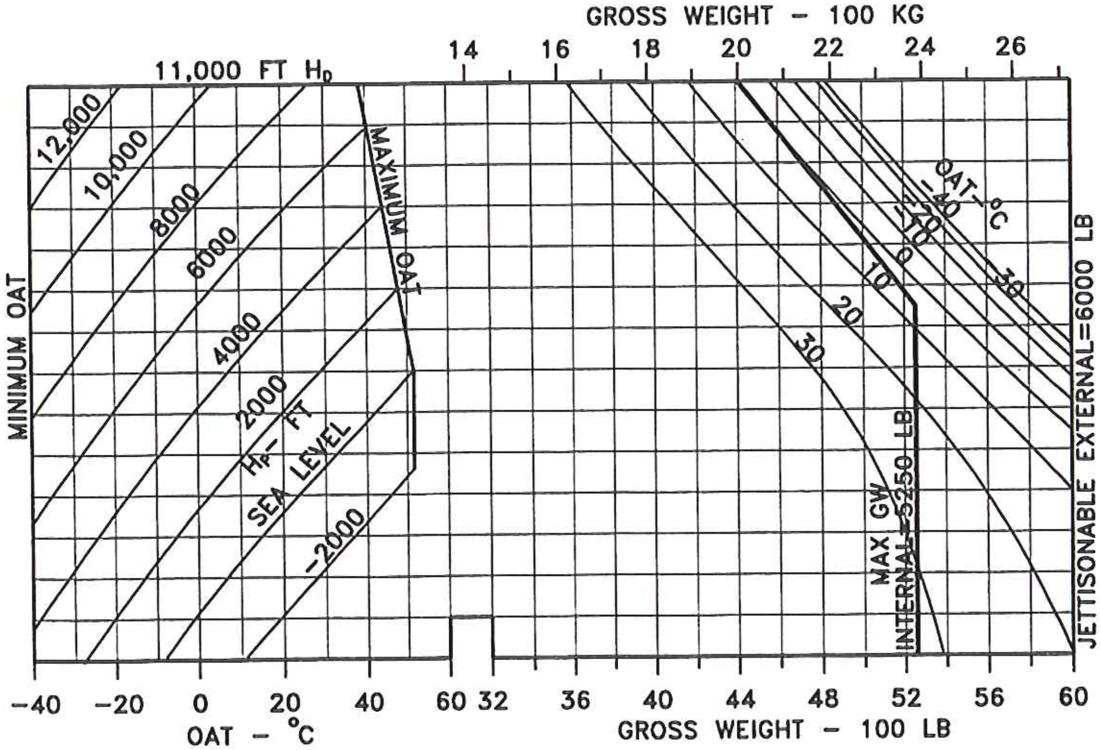
HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET
SNOW BAFFLES



407FS28-4-4-9

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 9 of 16)

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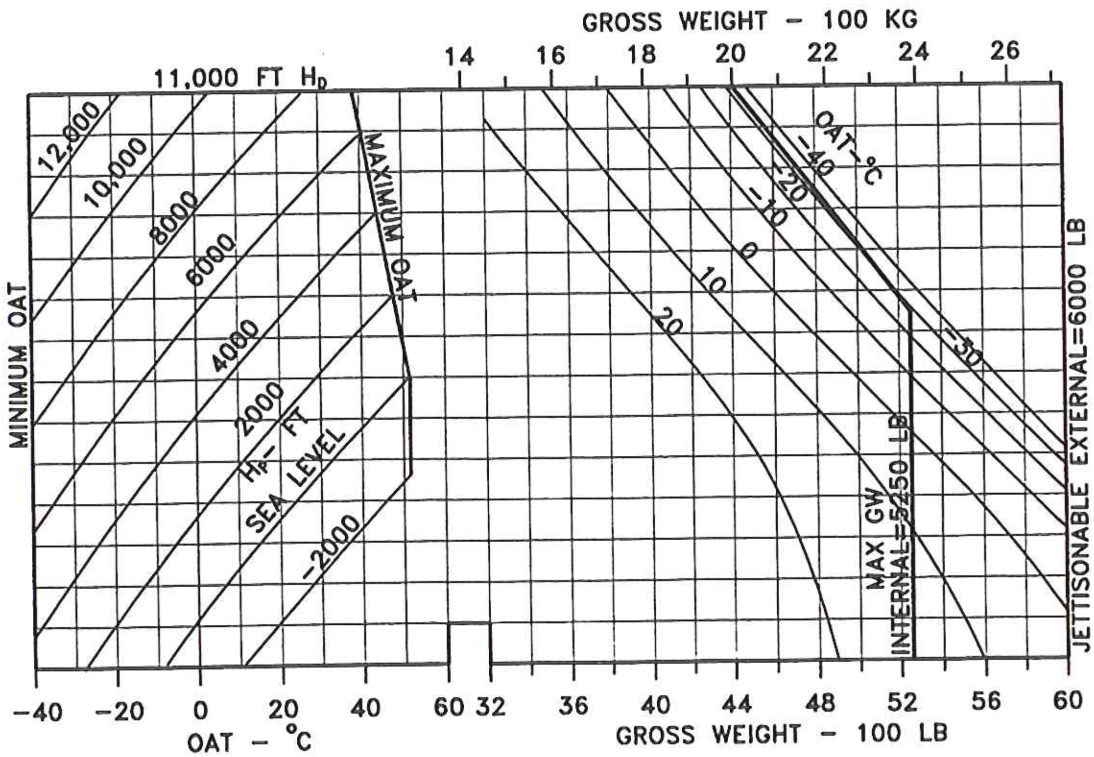
HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER ON
BASIC INLET
SNOW BAFFLES



407FS28-4-4-11

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 11 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET
SNOW BAFFLES

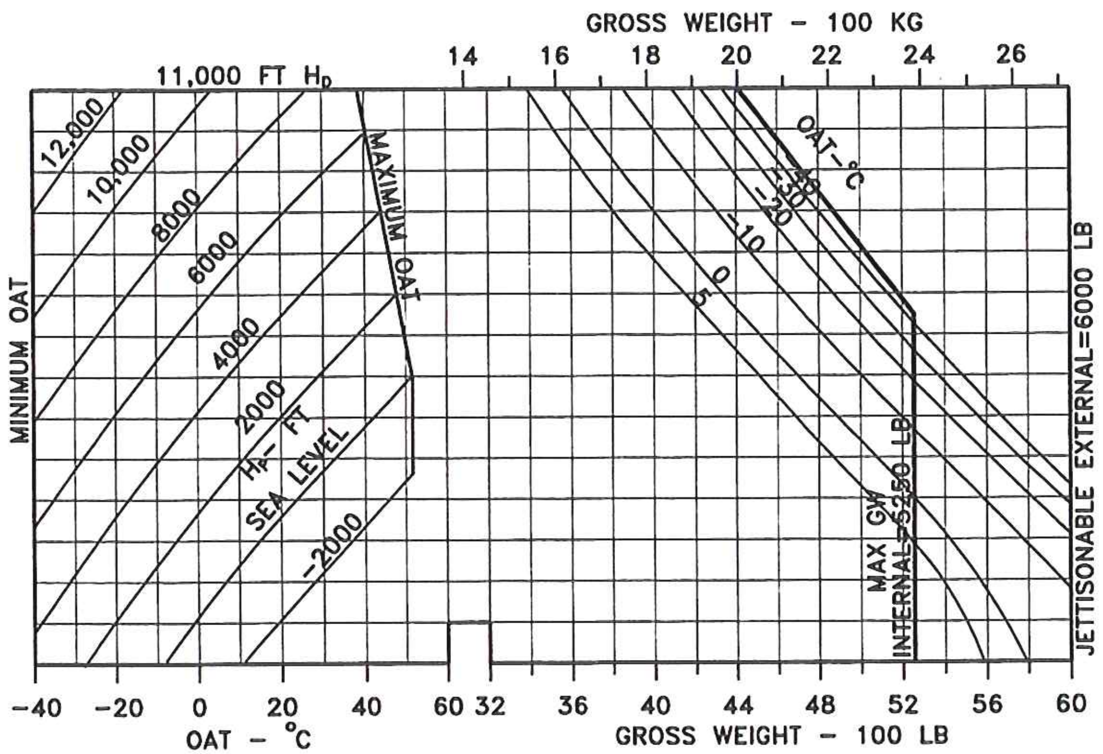

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 12 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR
SNOW BAFFLES

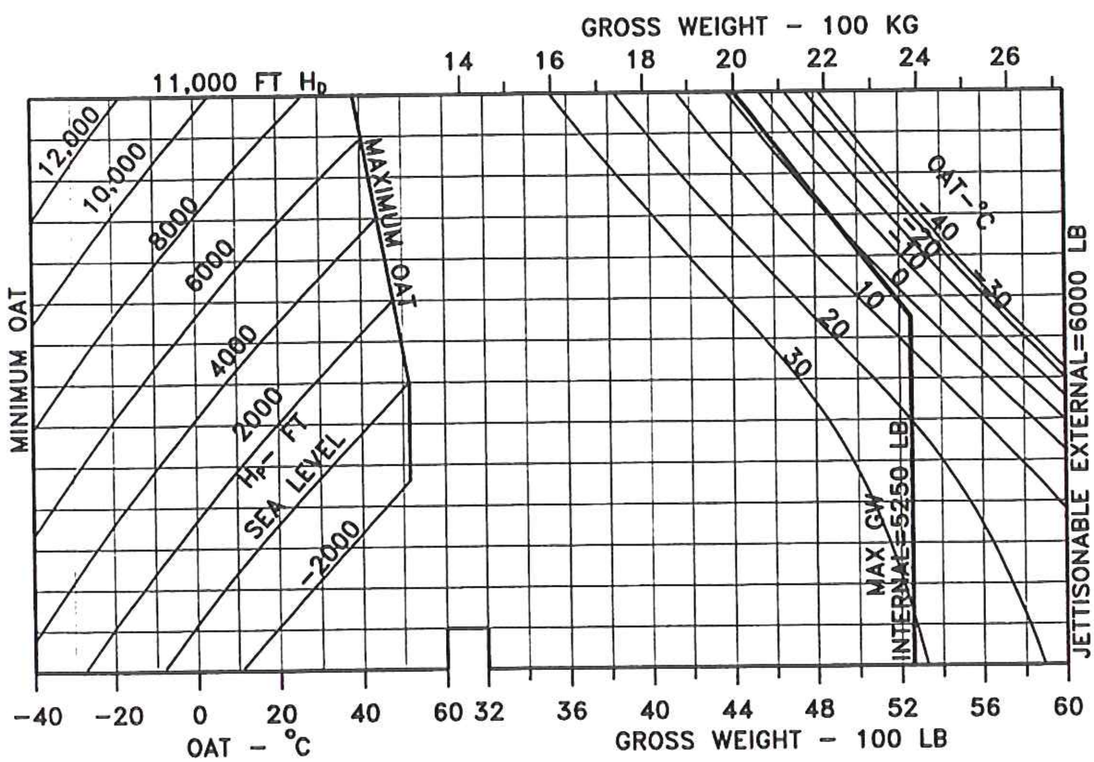

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 13 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
ANTI-ICE ON
PARTICLE SEPARATOR
SNOW BAFFLES

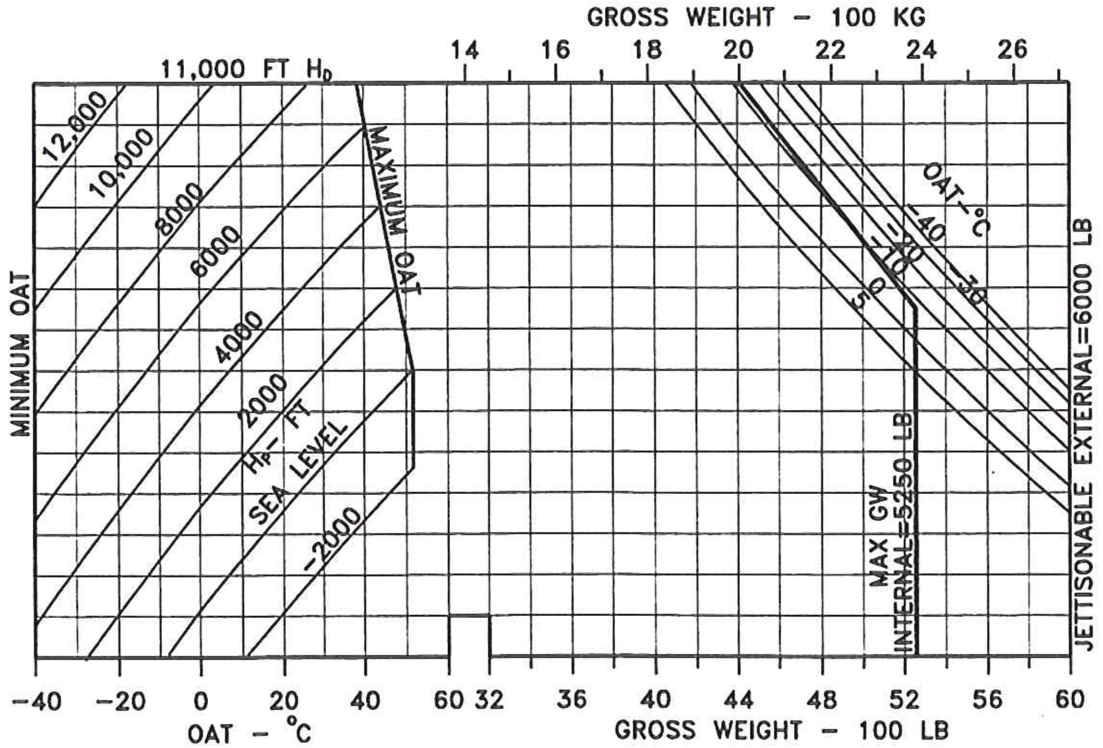

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 14 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET

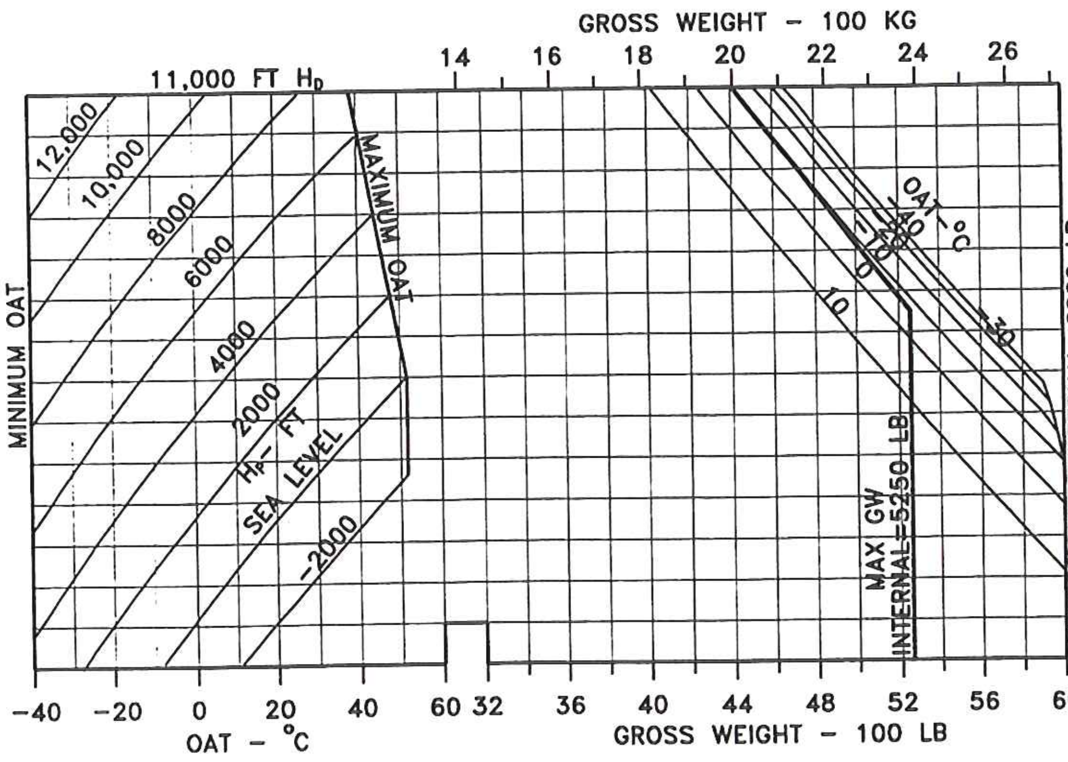

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 4 of 16)

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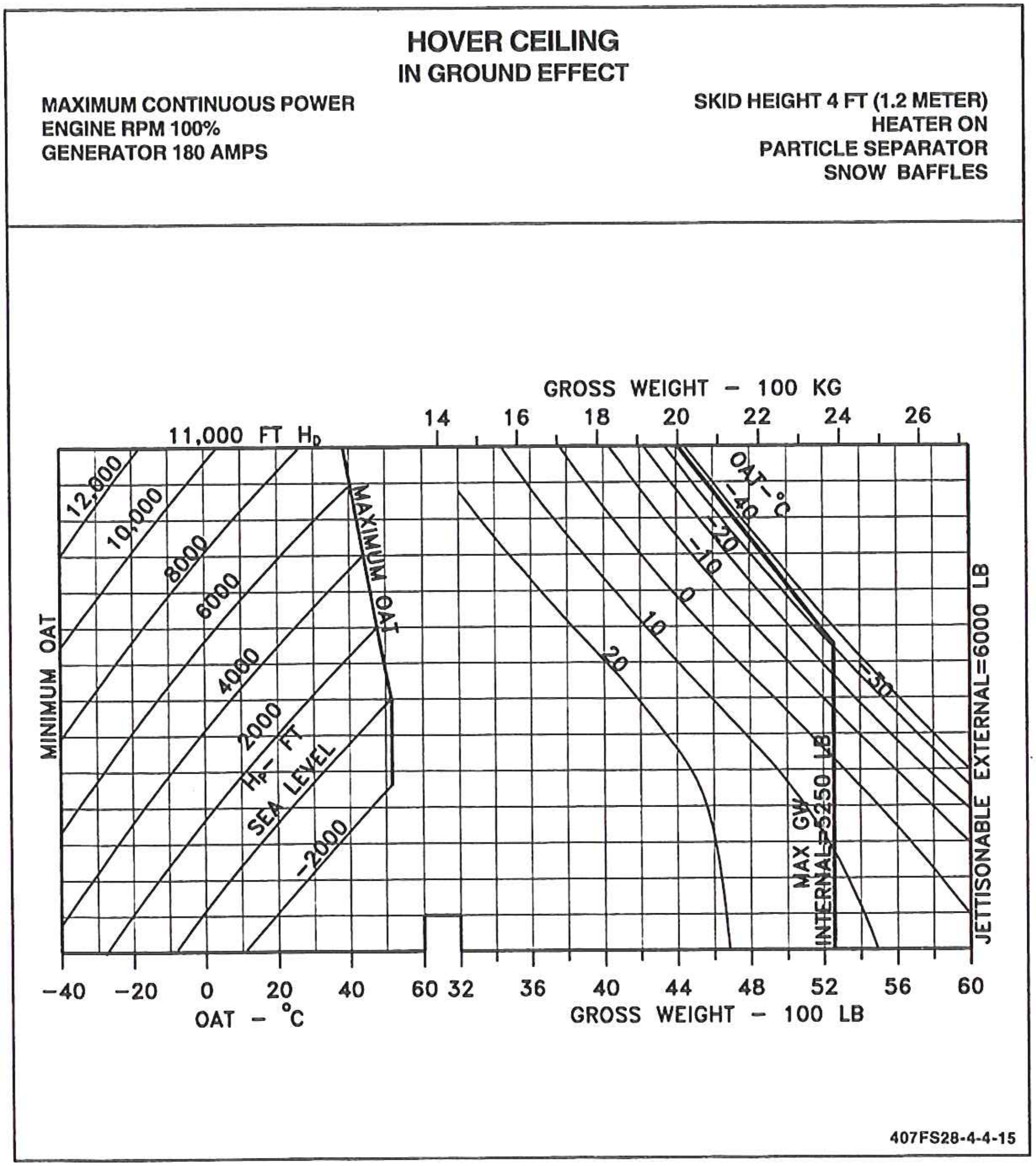

Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 15 of 16)

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HOVER CEILING
IN GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 4 FT (1.2 METER)
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR
SNOW BAFFLES

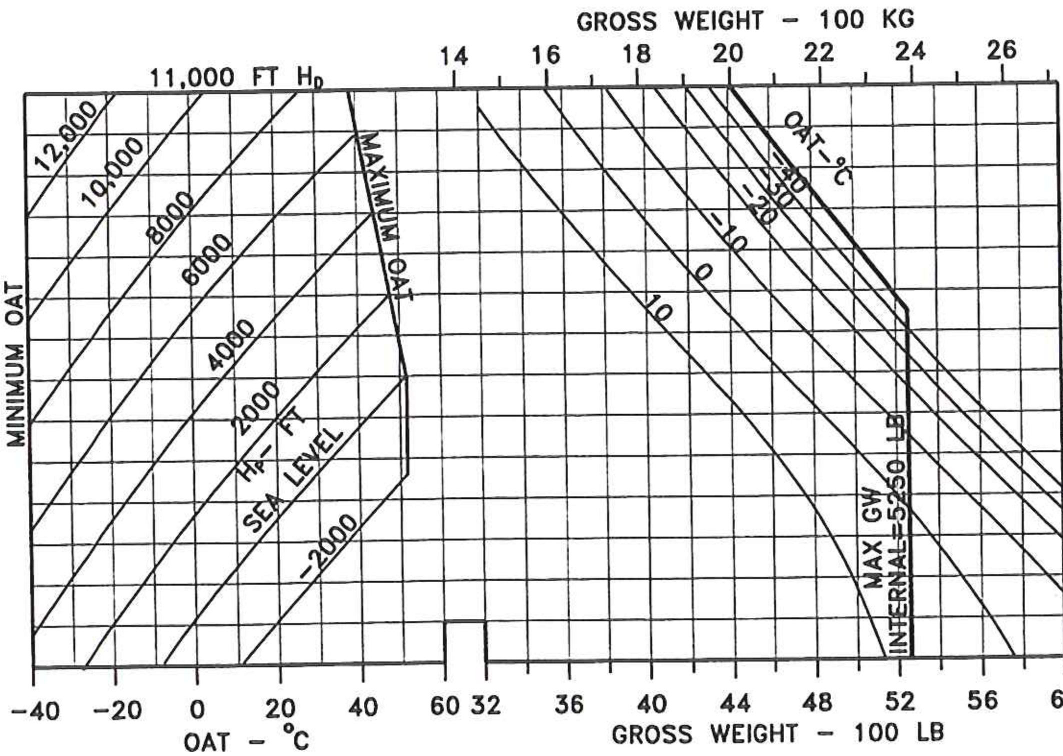

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Figure 4-4. Hover ceiling IGE - maximum continuous power (Sheet 16 of 16)

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HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET

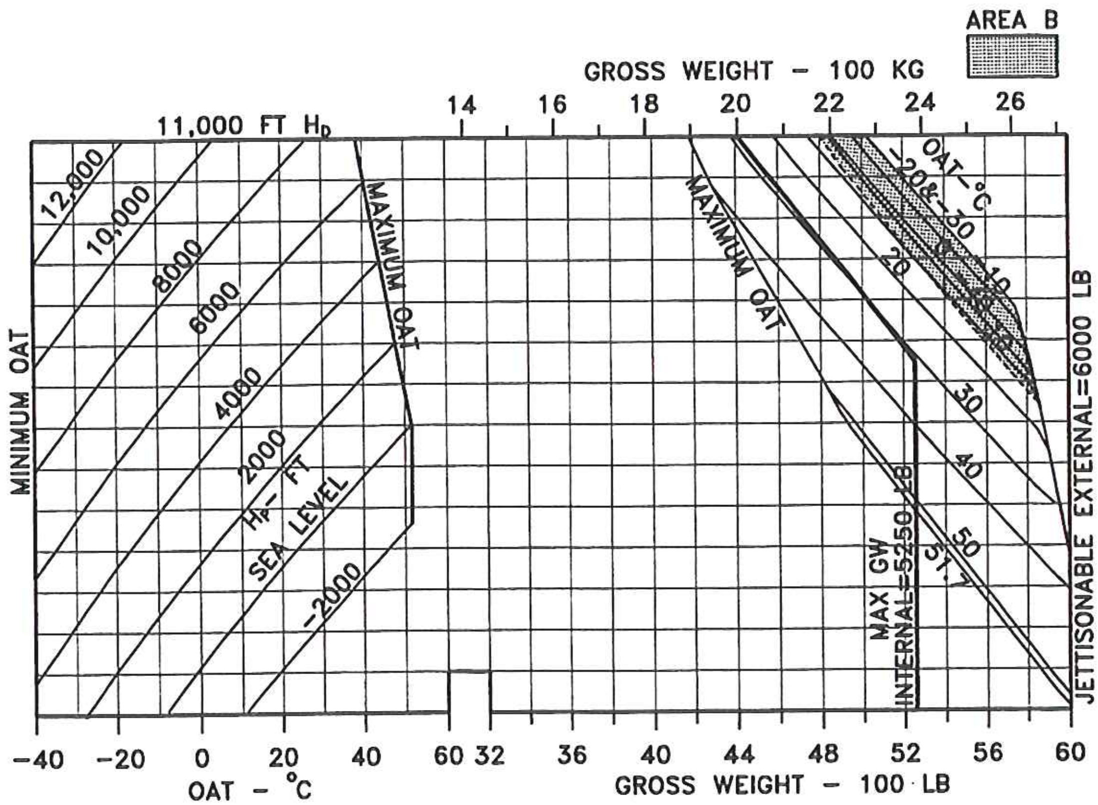

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Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 1 of 16)

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FOR TRAINING PURPOSES ONLY

HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
BASIC INLET

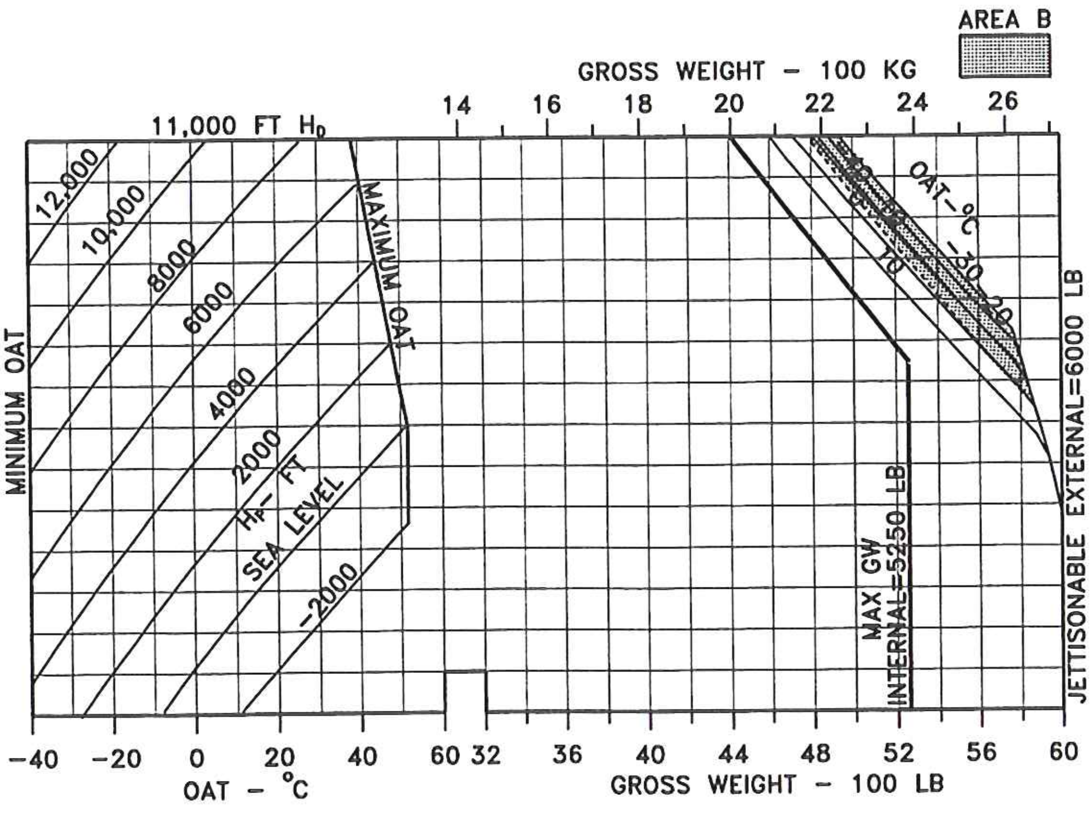

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 2 of 16)

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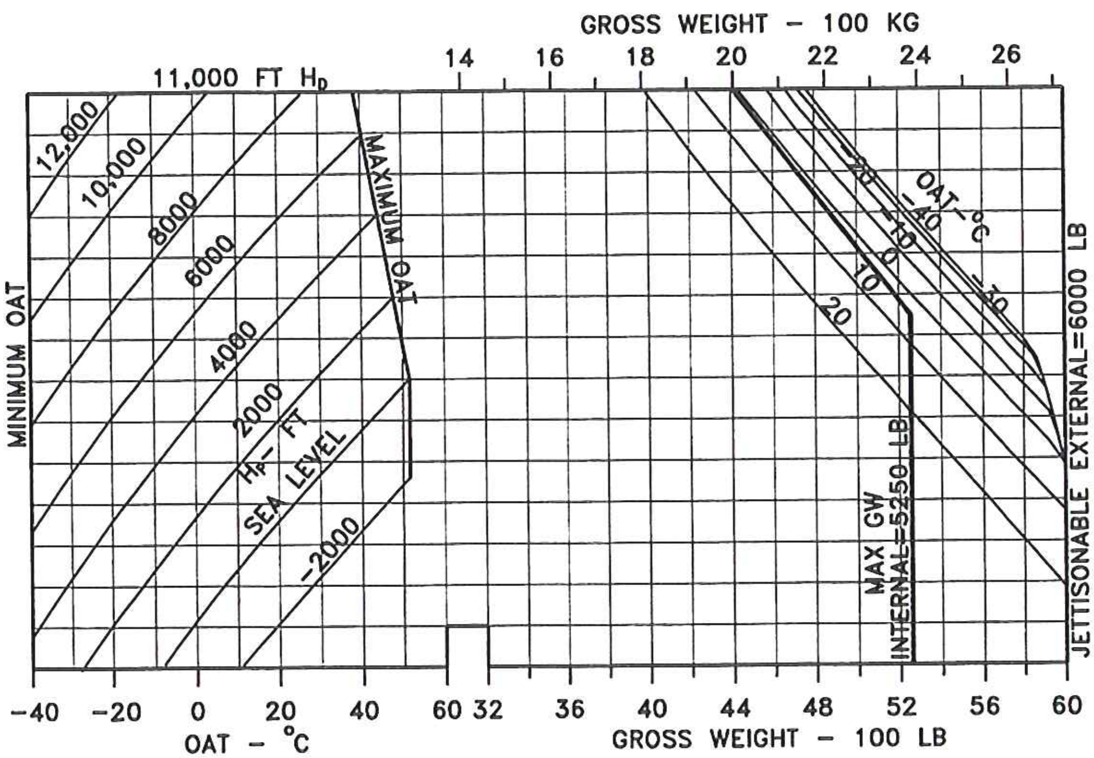
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
BASIC INLET



407FS28-4-5-3

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 3 of 16)

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HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR

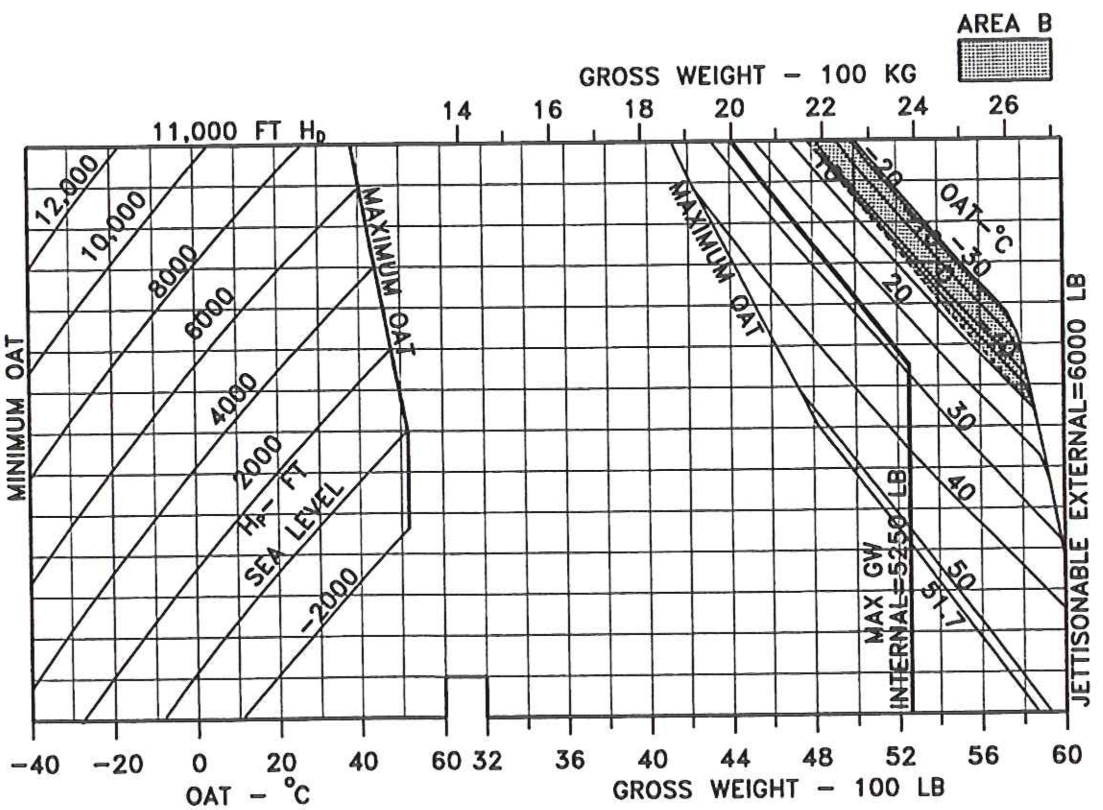

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 5 of 16)

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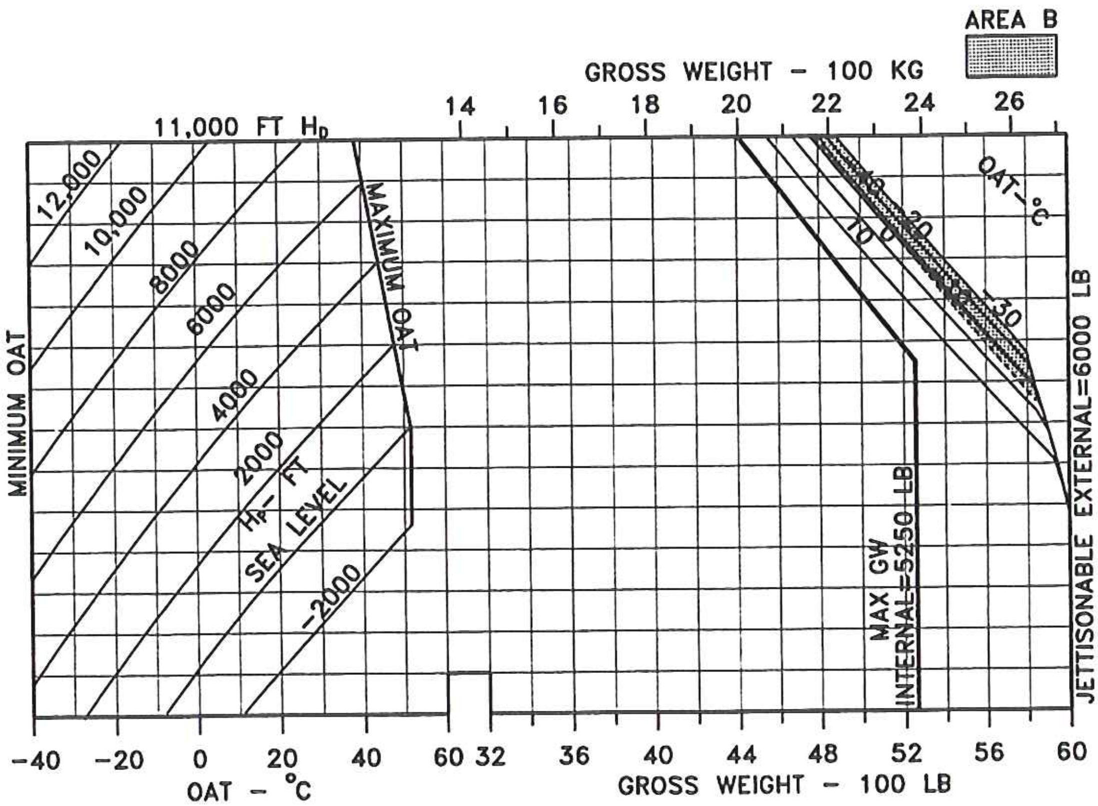
HOVER CEILING

OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
PARTICLE SEPARATOR



407FS28-4-5-6

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 6 of 16)

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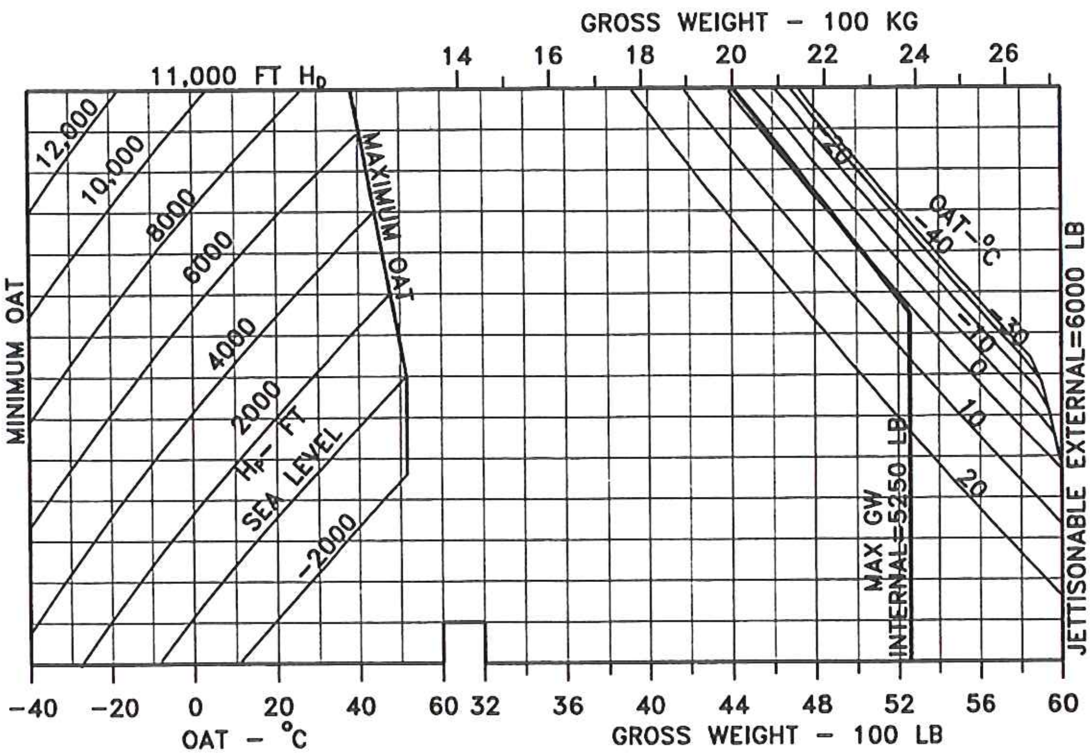
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
PARTICLE SEPARATOR



407FS28-4-5-7

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 7 of 16)

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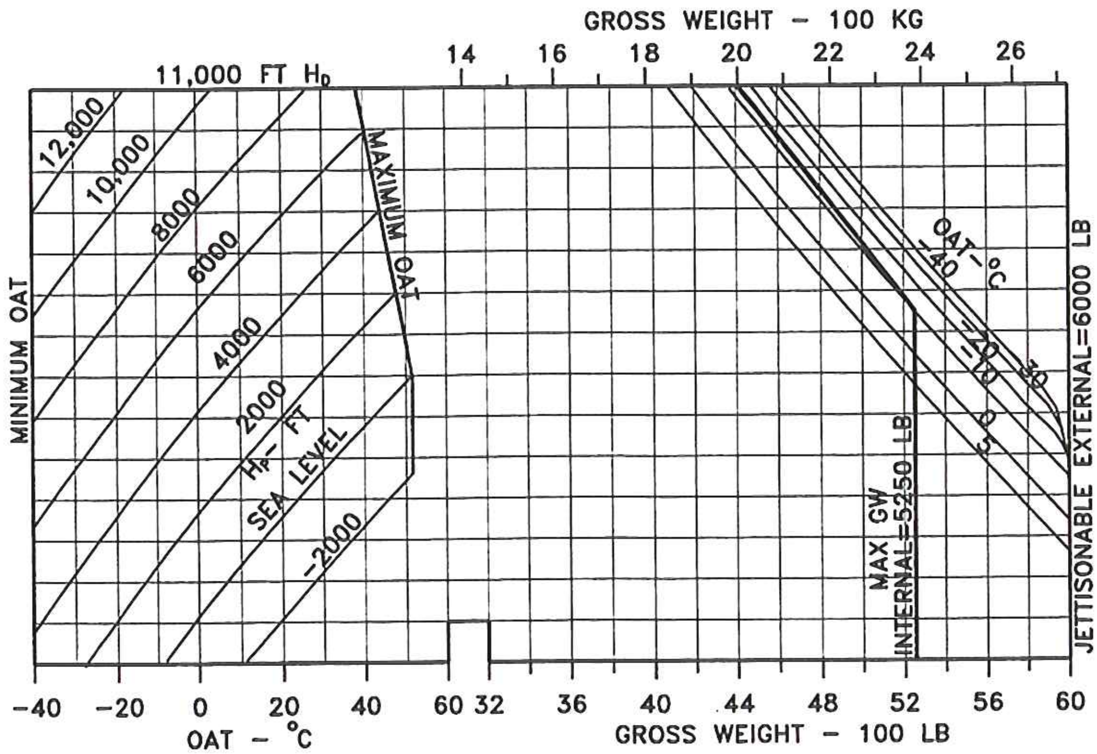
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR



407FS28-4-5-8

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 8 of 16)

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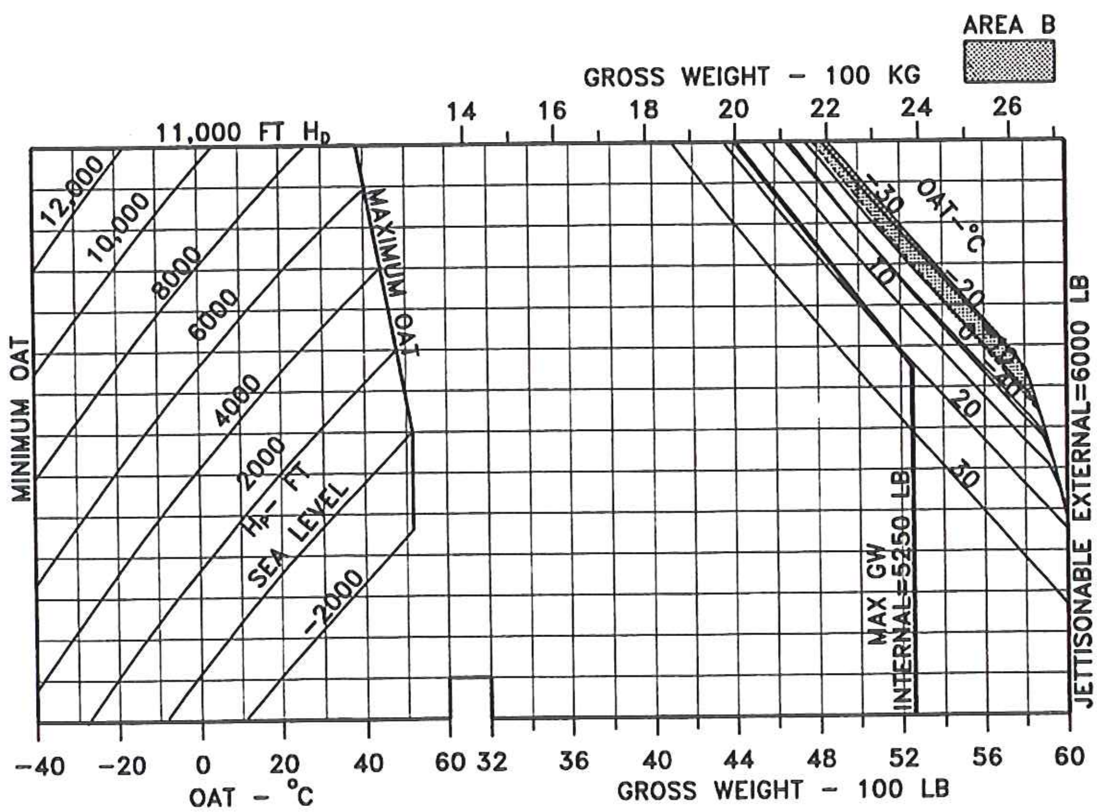
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET
SNOW BAFFLES



407FS28-4-5-9

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 9 of 16)

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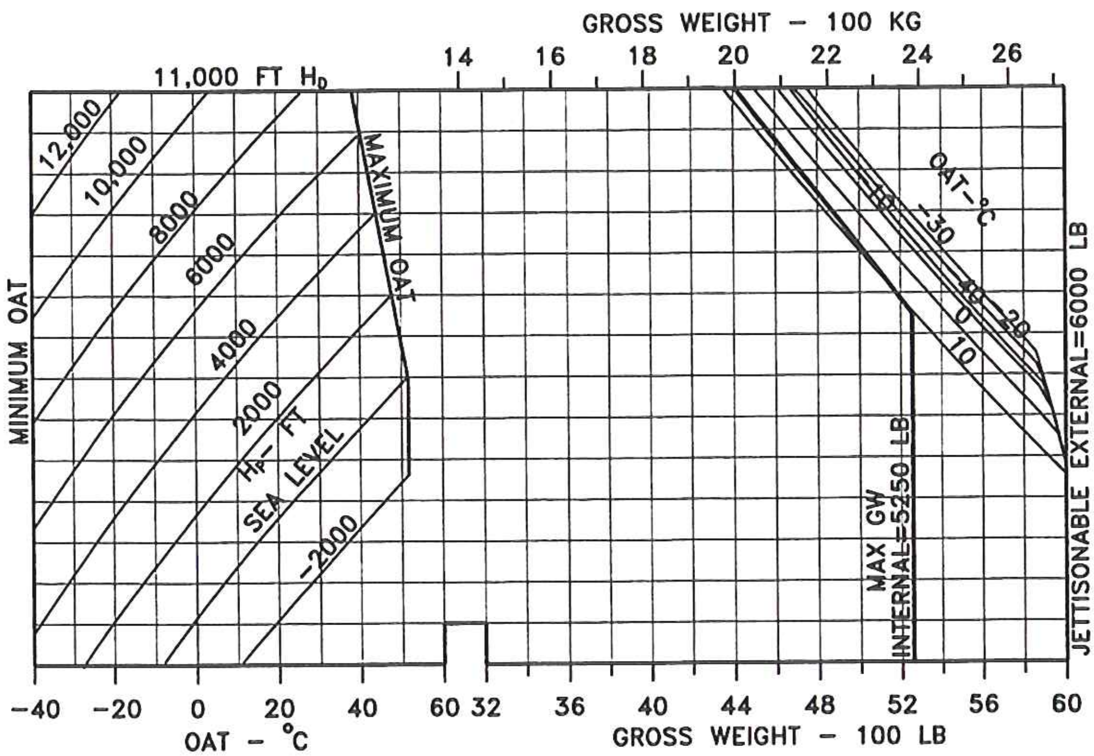
HOVER CEILING

OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
BASIC INLET
SNOW BAFFLES



407FS28-4-5-10

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 10 of 16)

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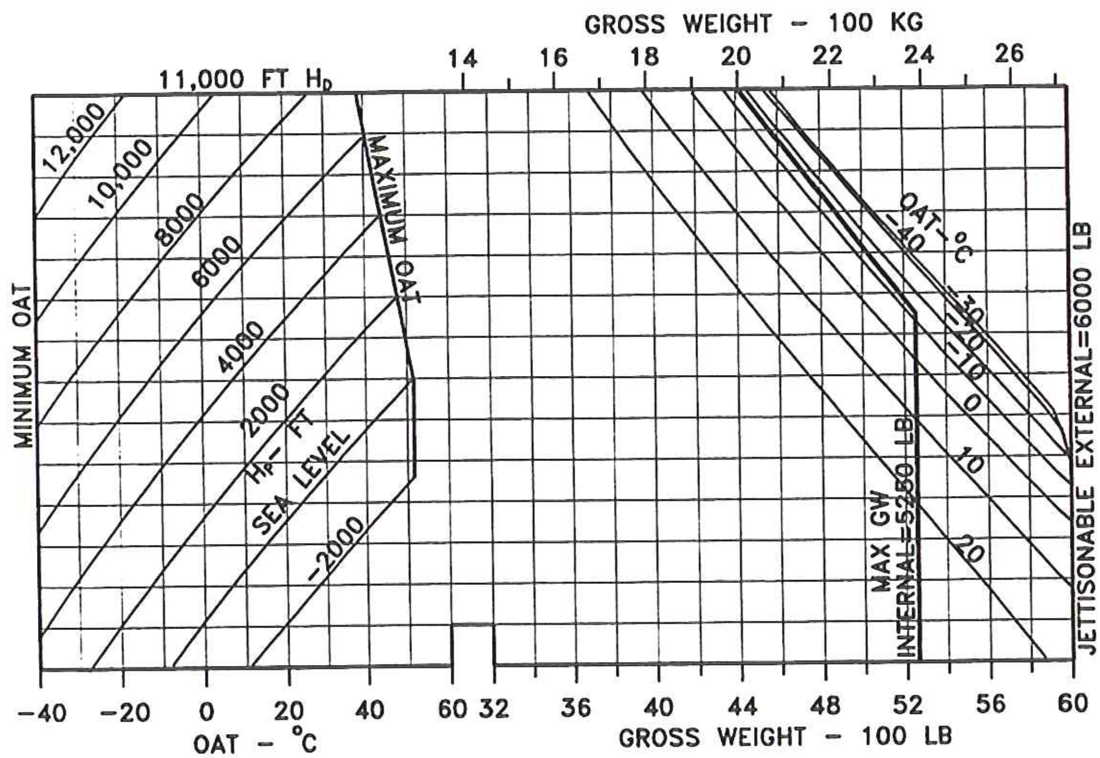
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
BASIC INLET
SNOW BAFFLES



407FS28-4-5-11

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 11 of 16)

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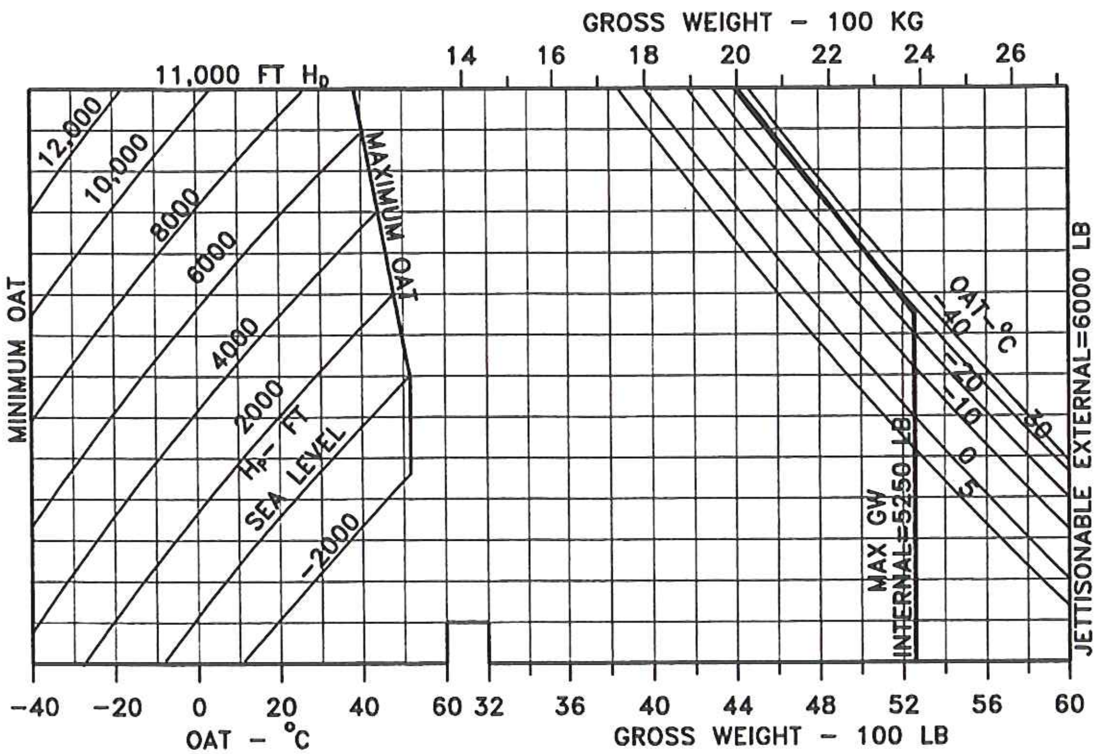
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET
SNOW BAFFLES



407FS28-4-5-12

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 12 of 16)

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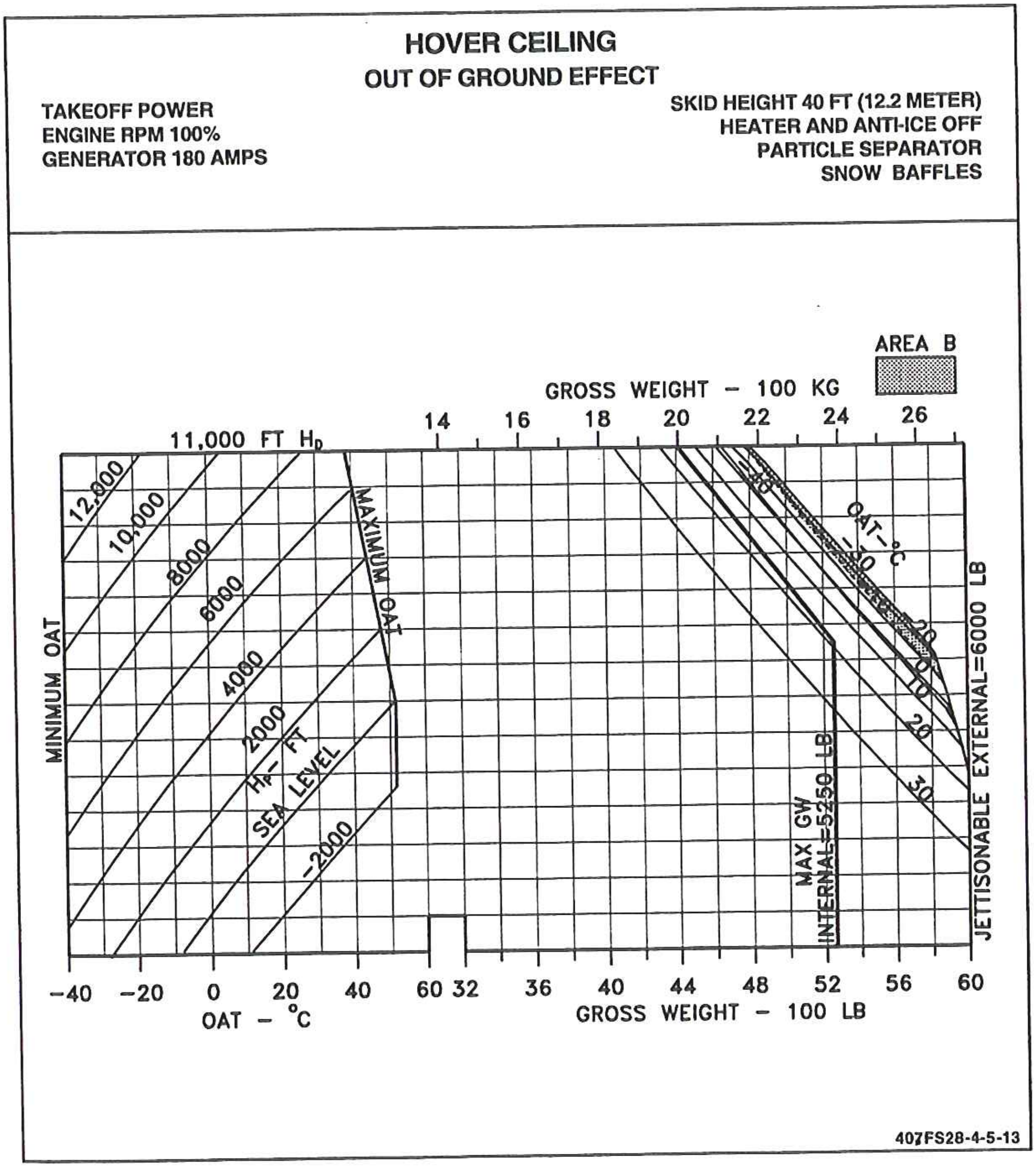

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 13 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
PARTICLE SEPARATOR
SNOW BAFFLES

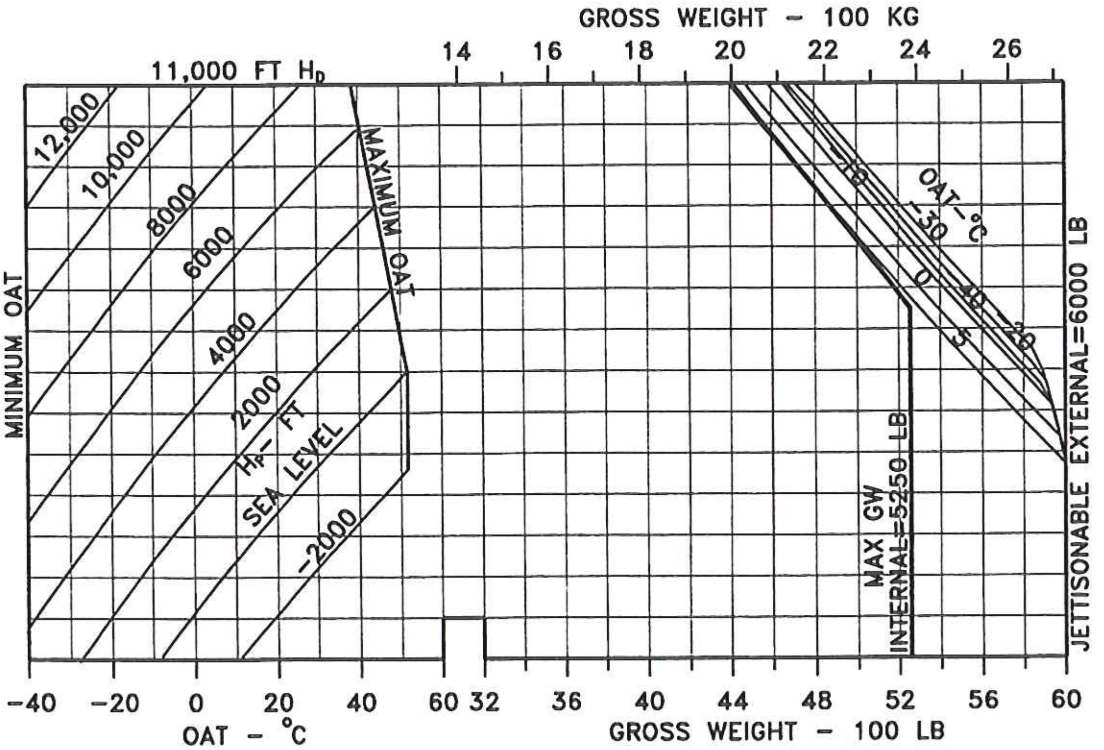

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Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 14 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
PARTICLE SEPARATOR
SNOW BAFFLES

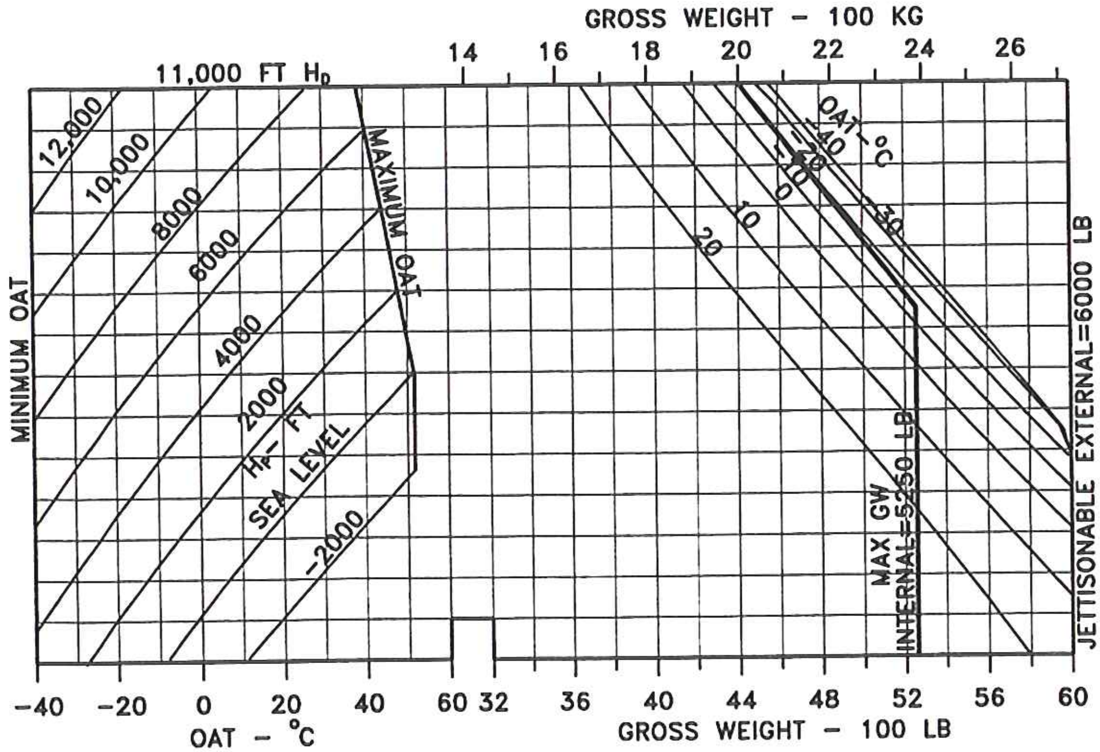

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 15 of 16)

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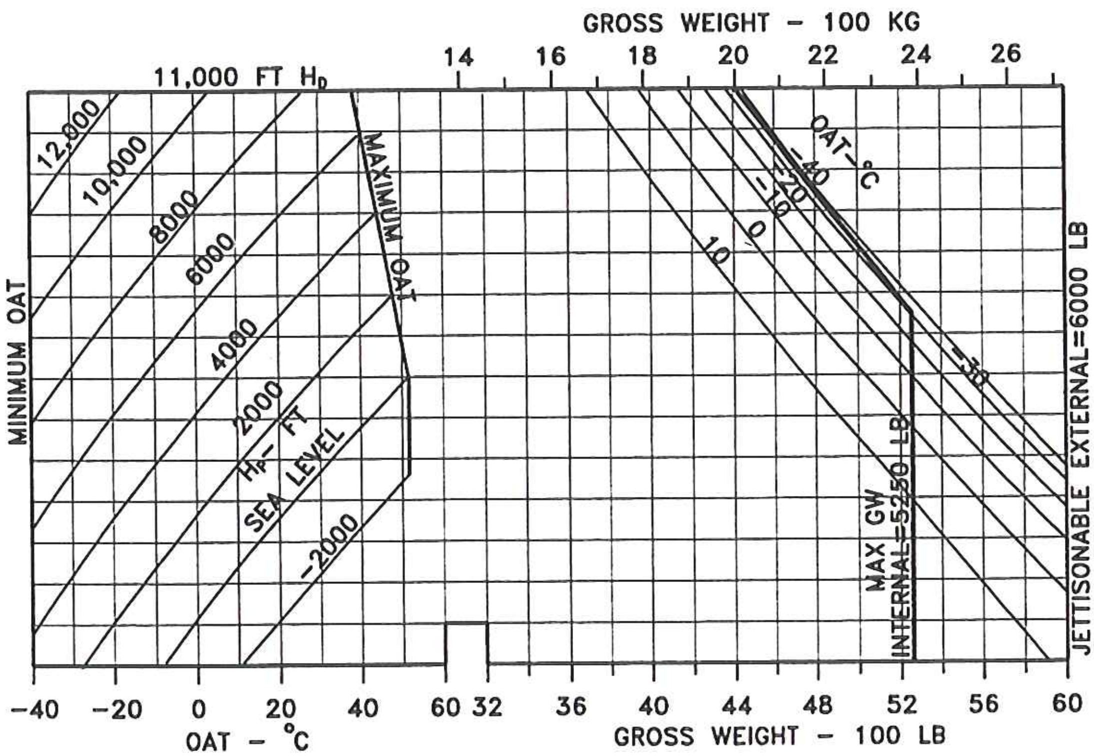
HOVER CEILING
OUT OF GROUND EFFECT

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR
SNOW BAFFLES



407FS28-4-5-16

Figure 4-5. Hover ceiling OGE - takeoff power (Sheet 16 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
BASIC INLET

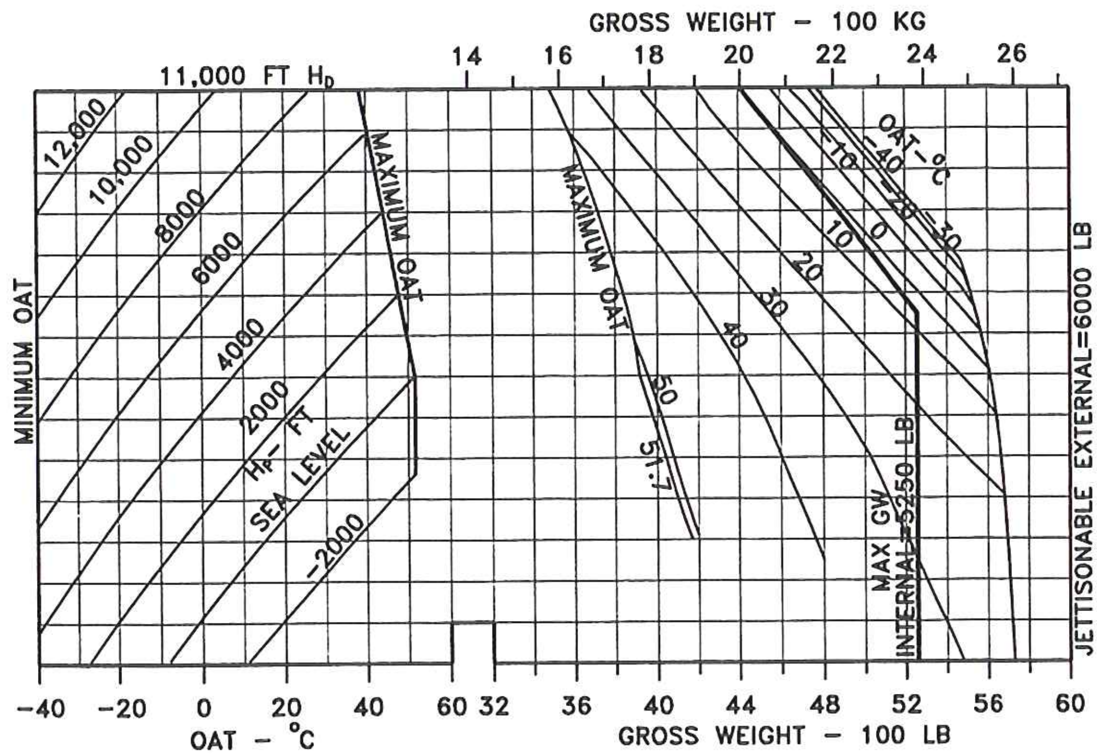

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 1 of 16)

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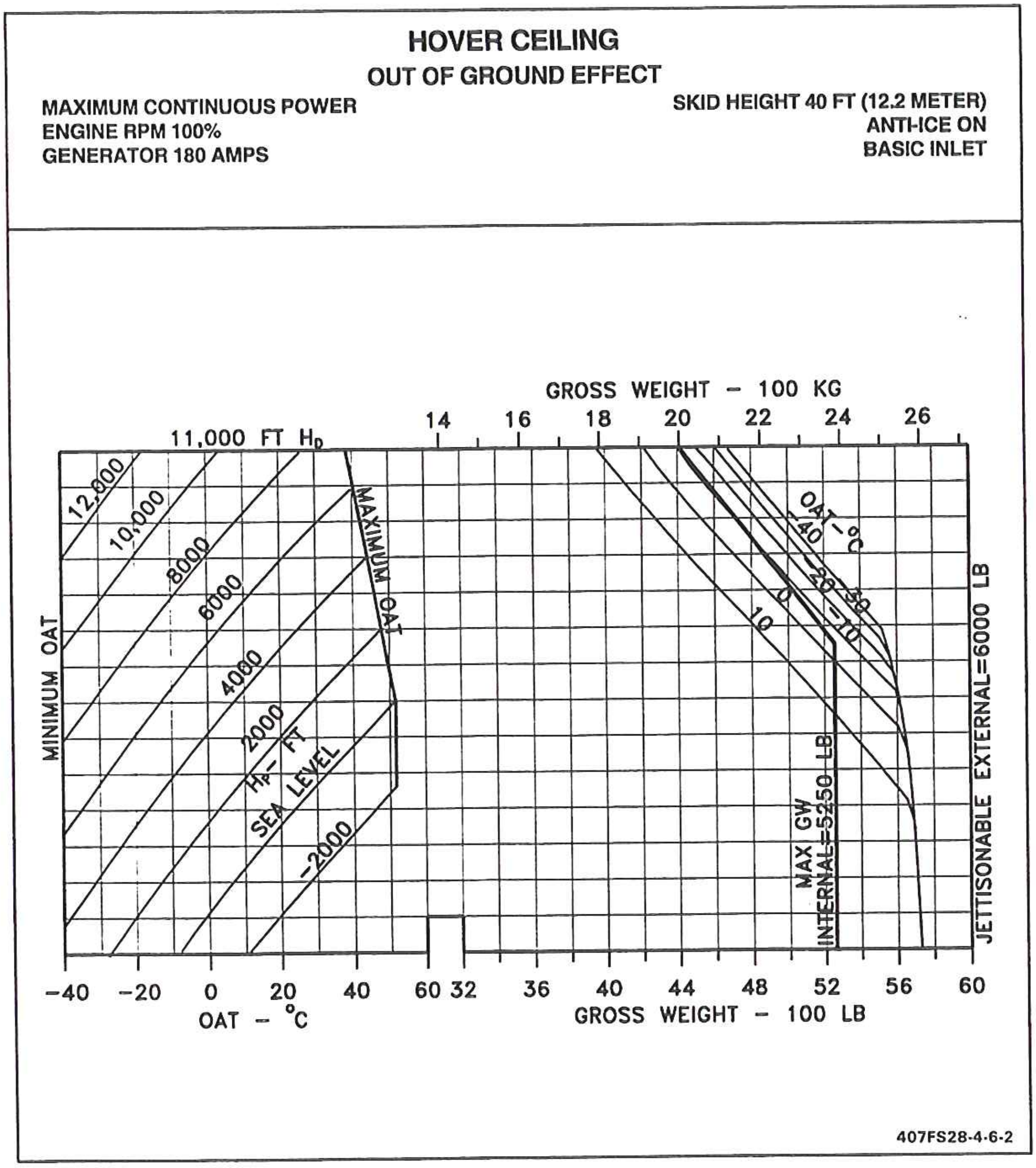

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 2 of 16)

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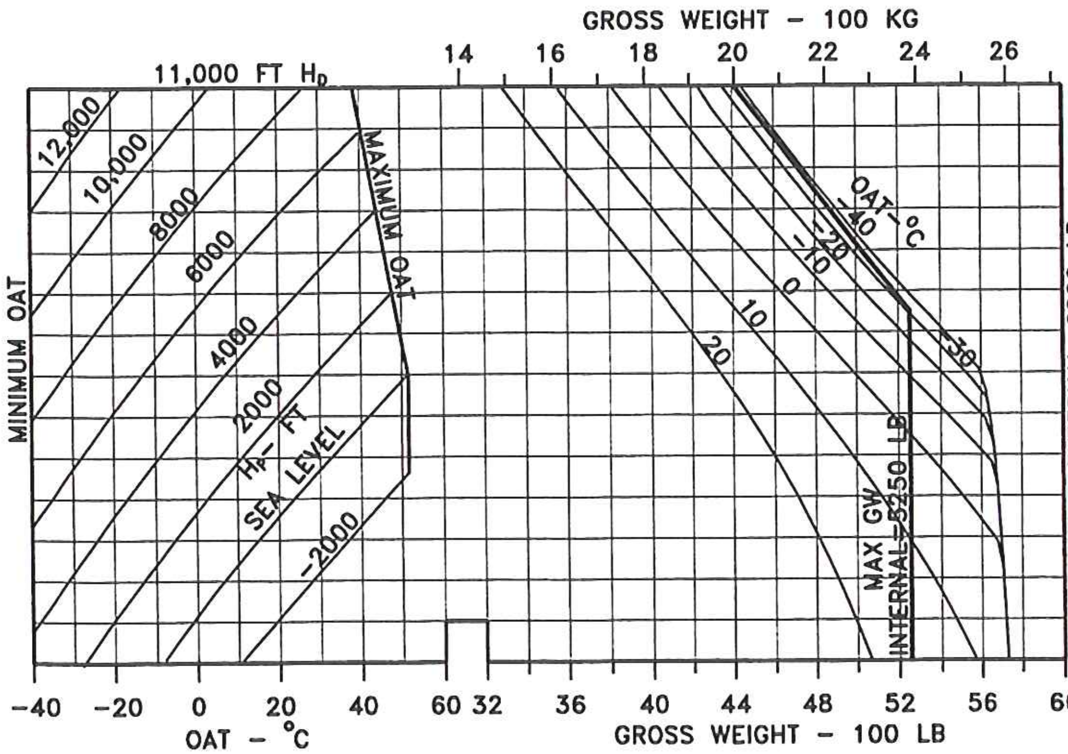
HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
BASIC INLET



407FS28-4-6-3

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 3 of 16)

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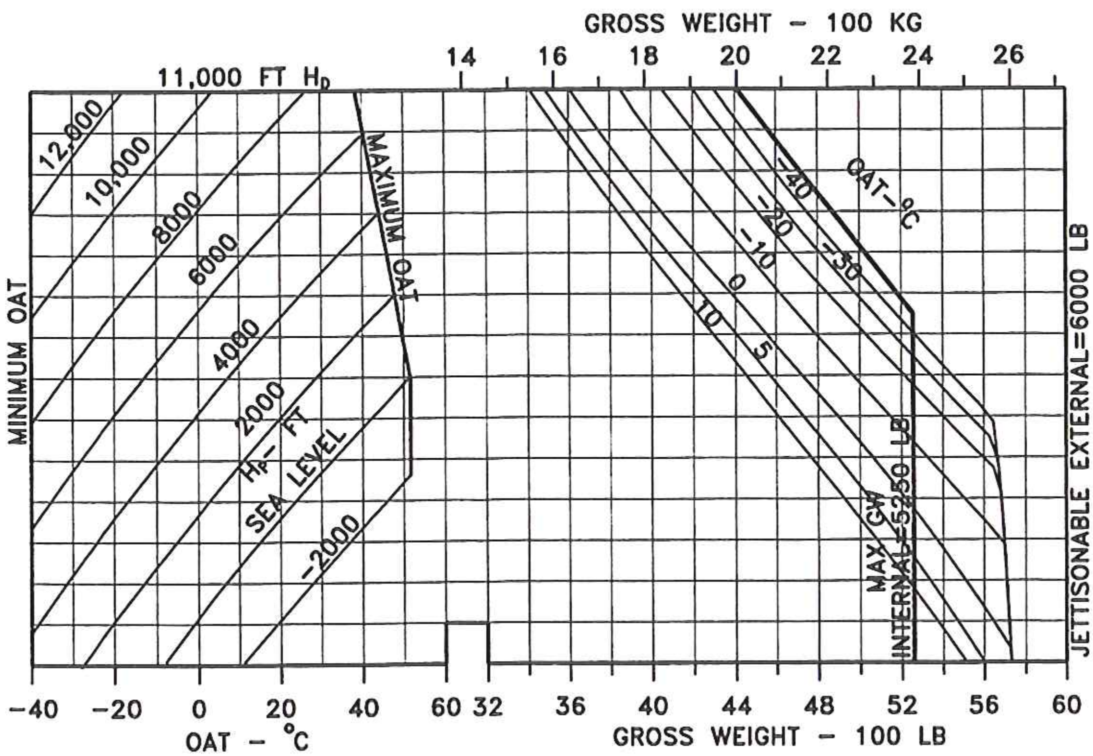
HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET



407FS28-4-6-4

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 4 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR

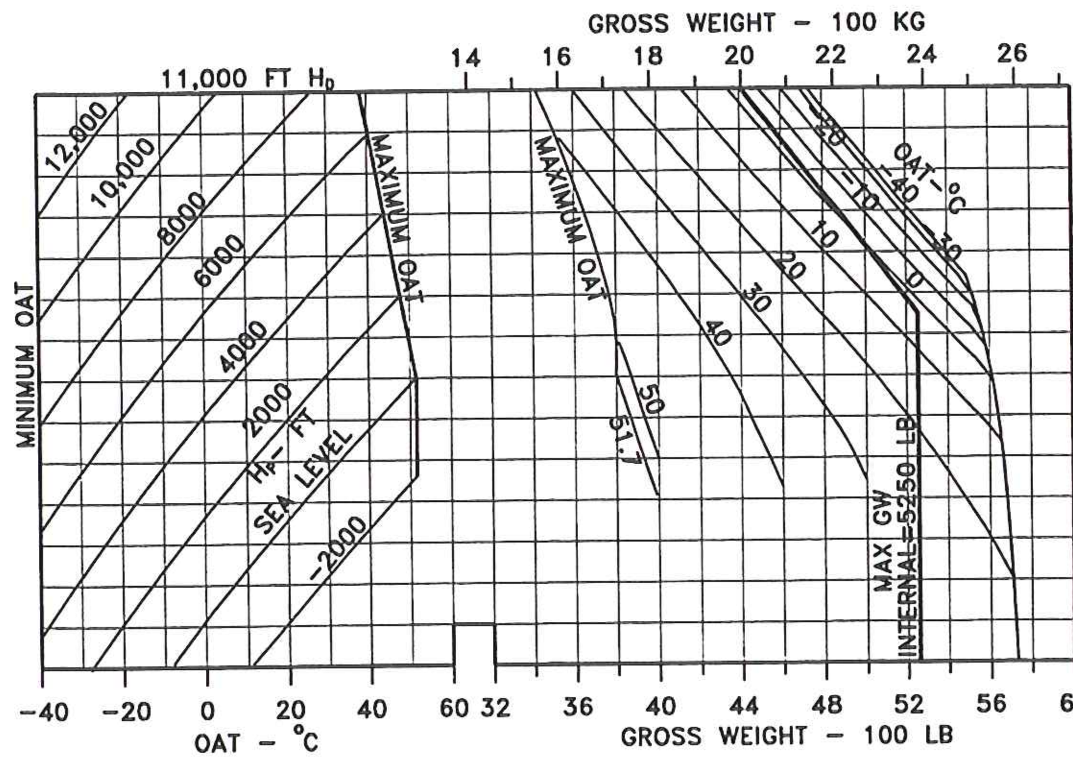

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Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 5 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
PARTICLE SEPARATOR

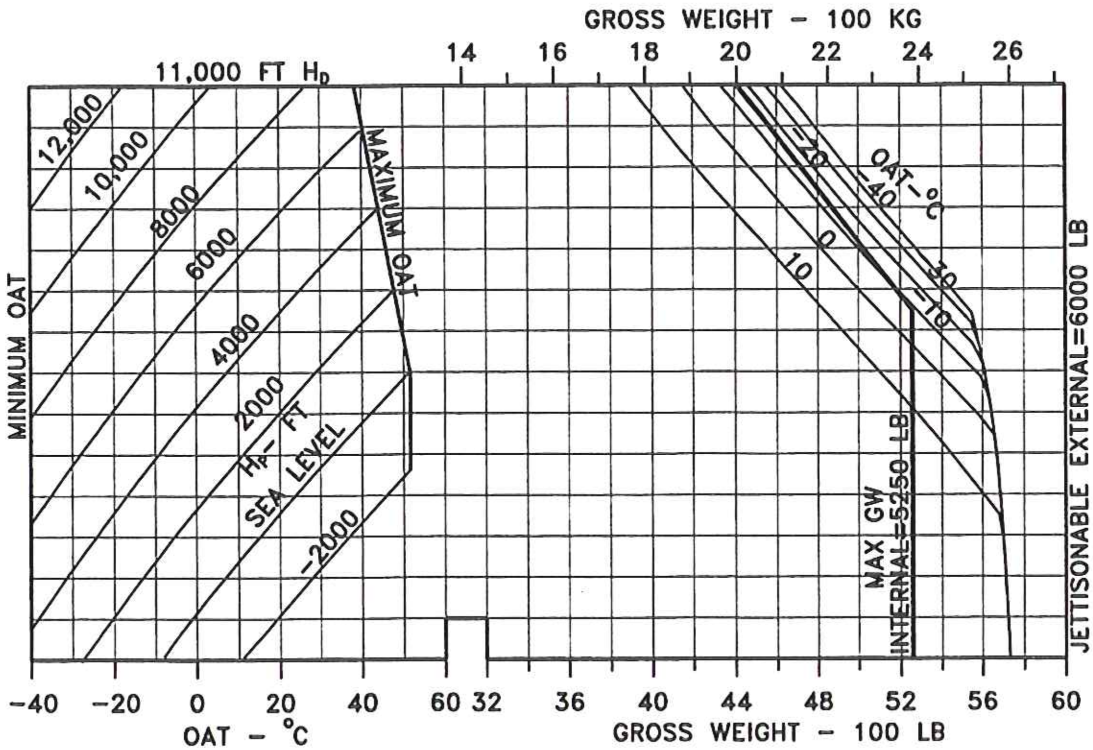

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Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 6 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
PARTICLE SEPARATOR

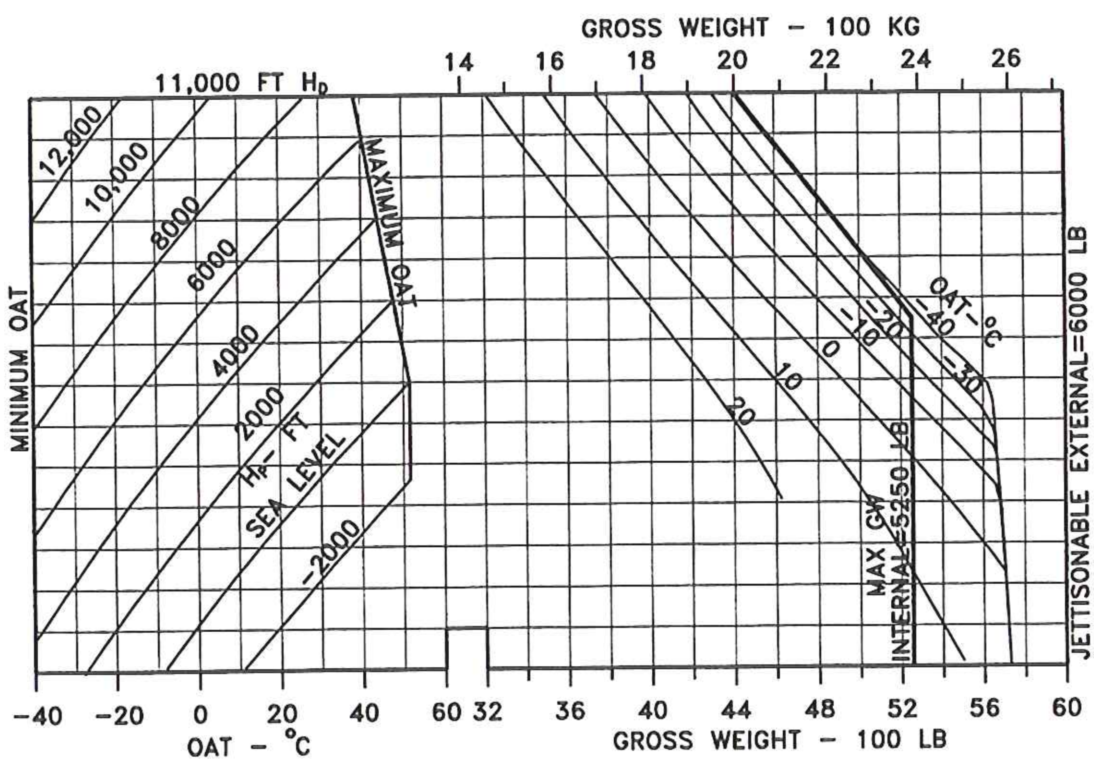

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 7 of 16)

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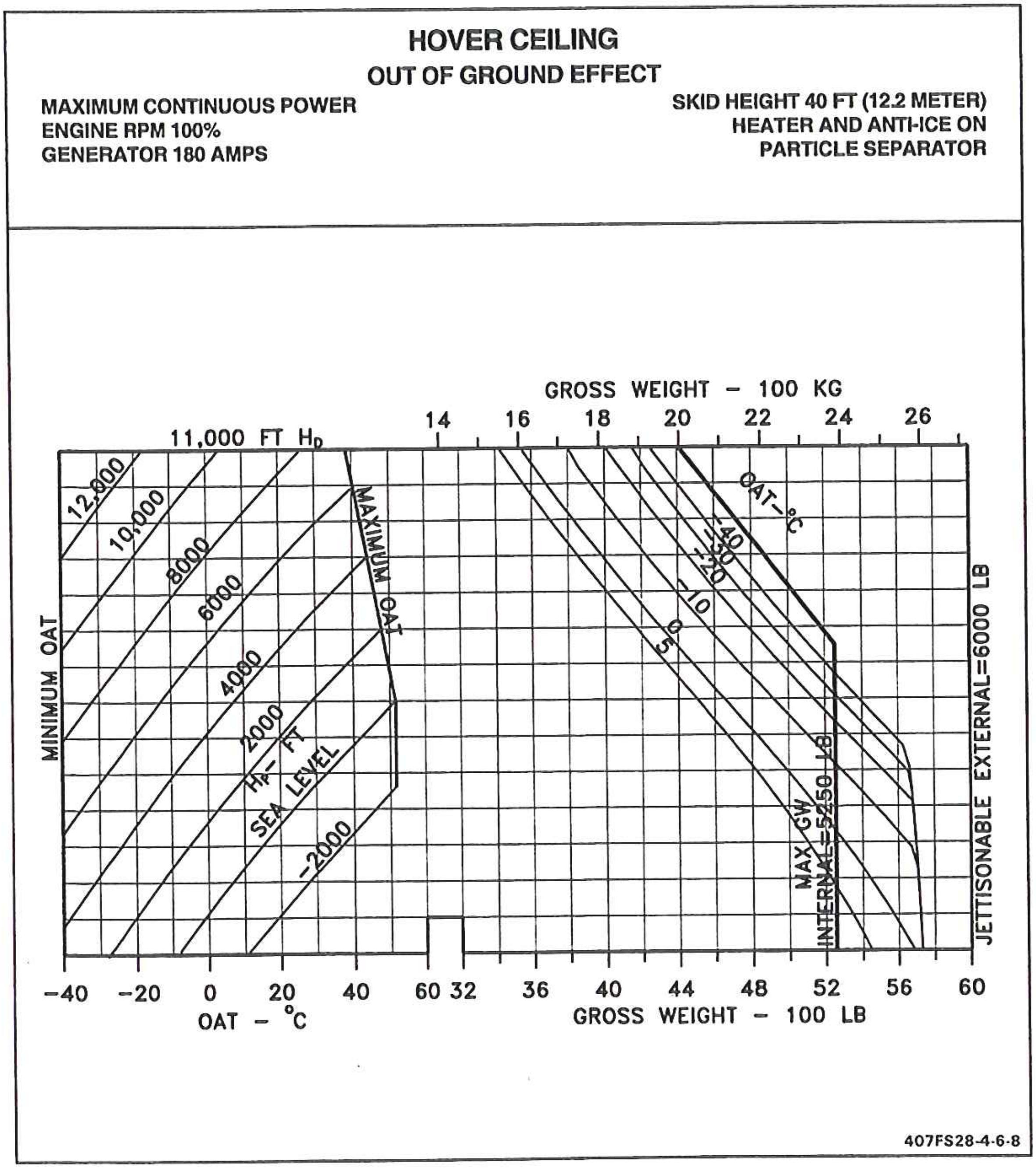

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 8 of 16)

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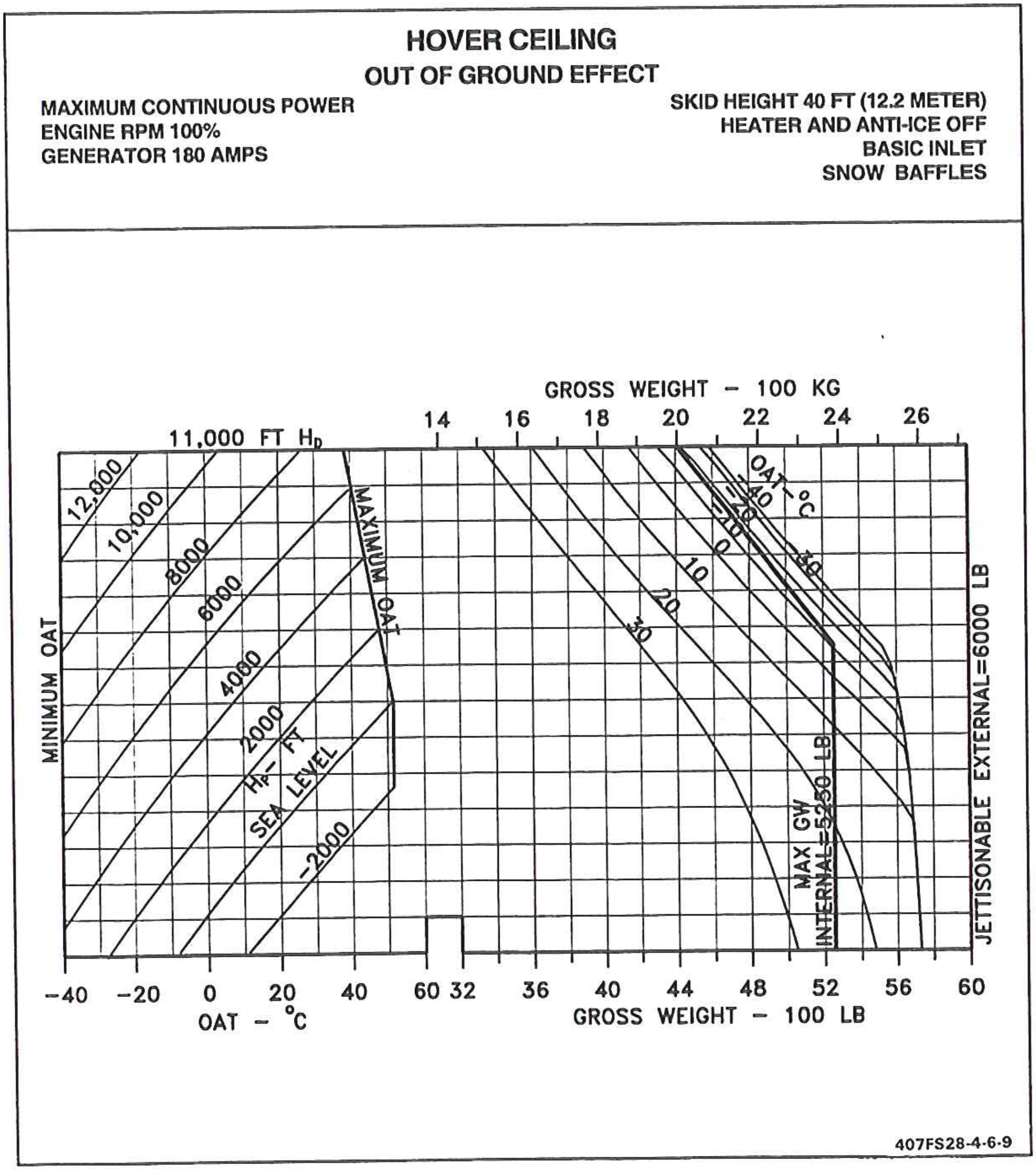

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 9 of 16)

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HOVER CEILING
OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
BASIC INLET
SNOW BAFFLES

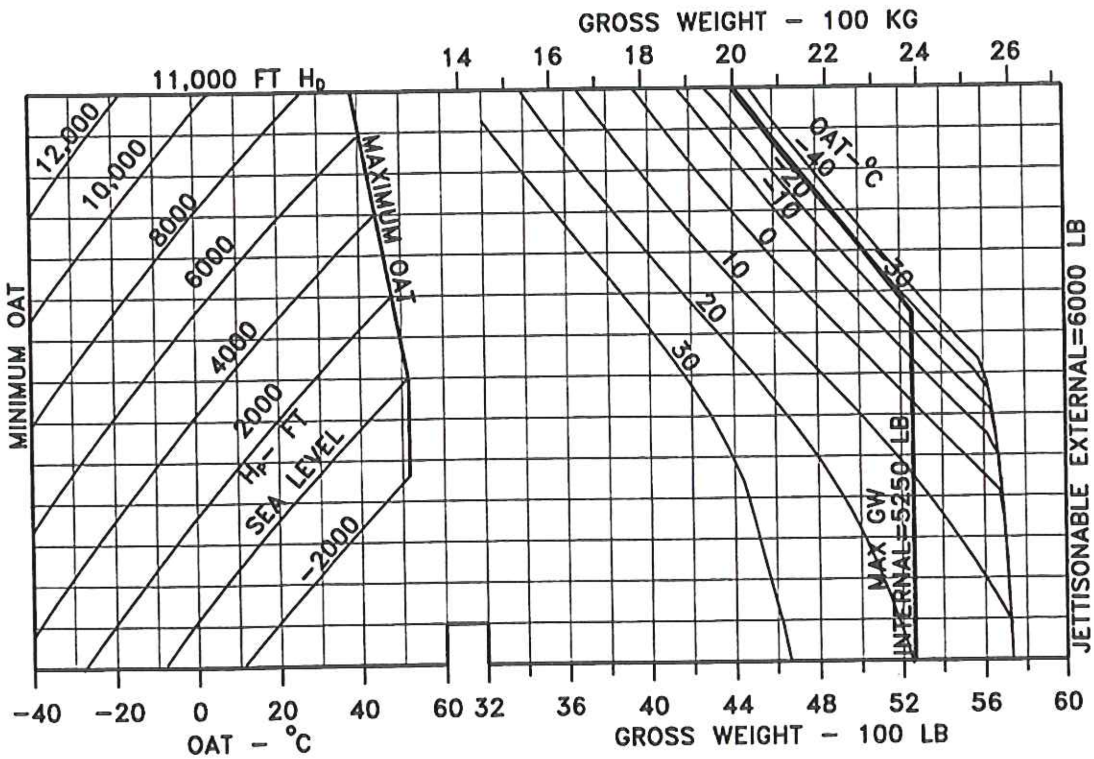

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Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 10 of 16)

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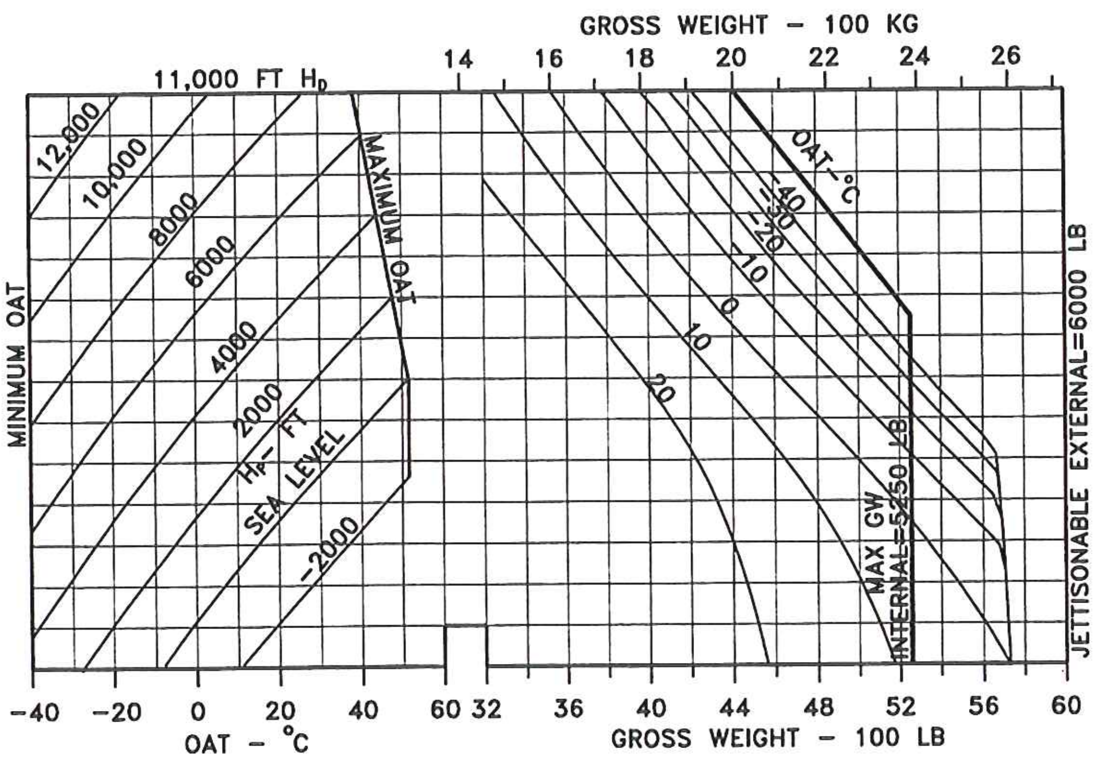
HOVER CEILING
OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
BASIC INLET
SNOW BAFFLES



407FS28-4-6-11

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 11 of 16)

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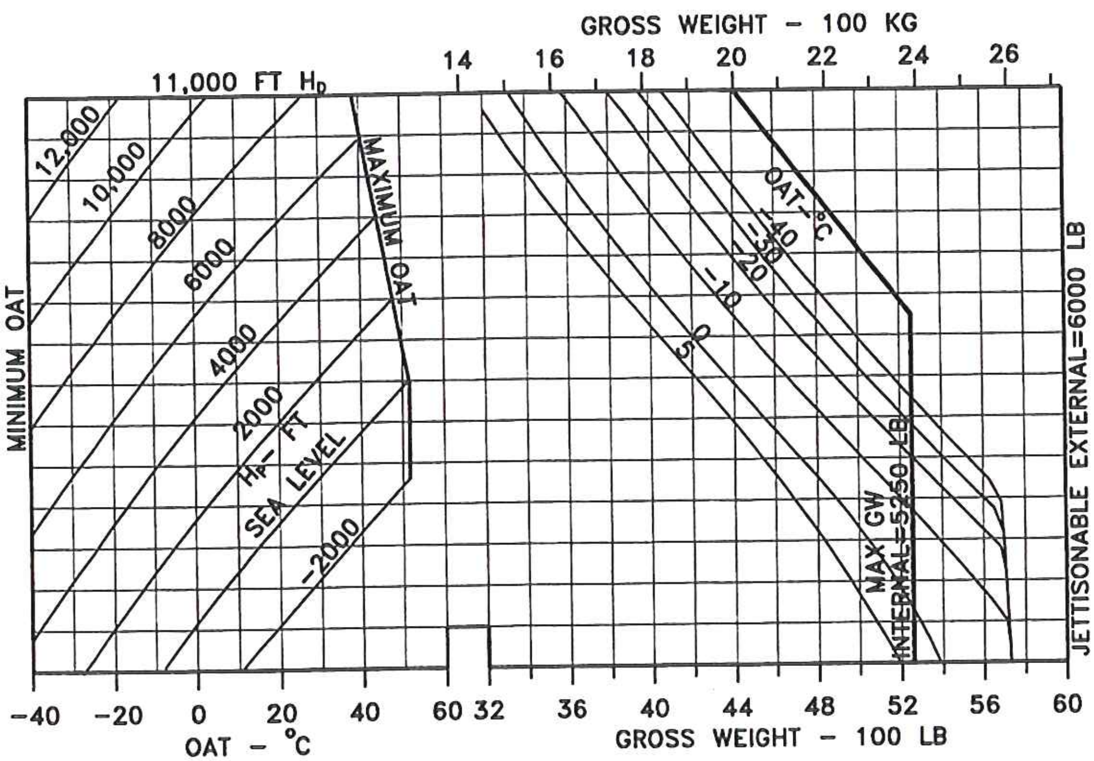
HOVER CEILING
OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
BASIC INLET
SNOW BAFFLES



407FS28-4-6-12

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 12 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR
SNOW BAFFLES

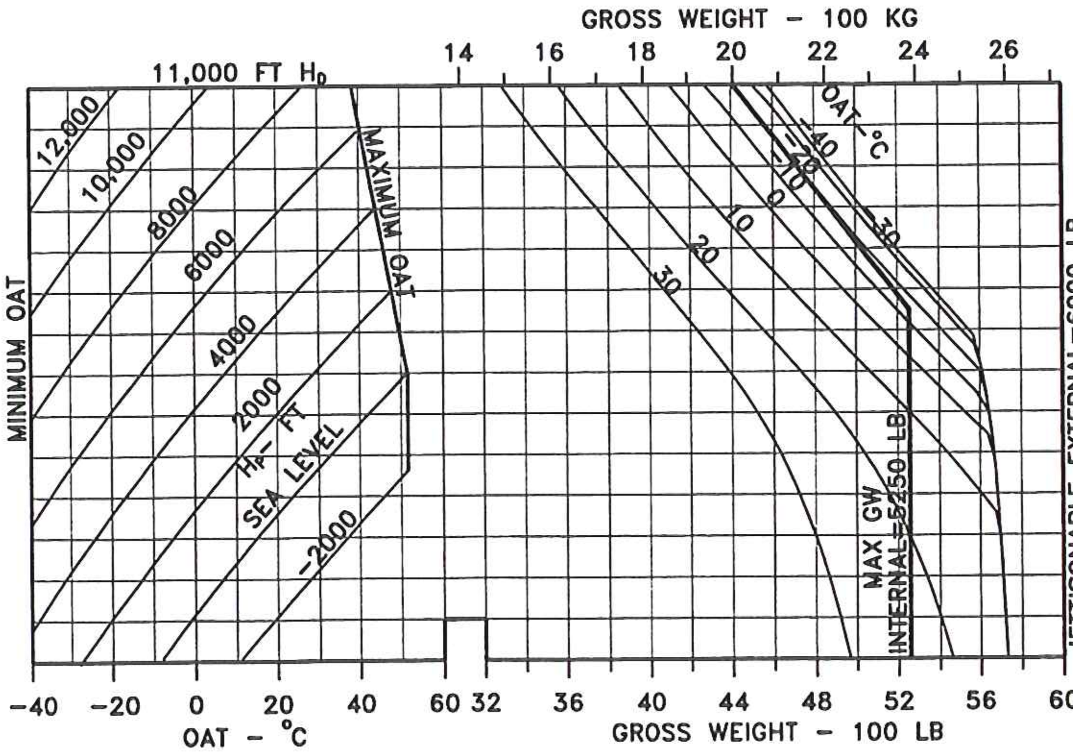

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 13 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
ANTI-ICE ON
PARTICLE SEPARATOR
SNOW BAFFLES

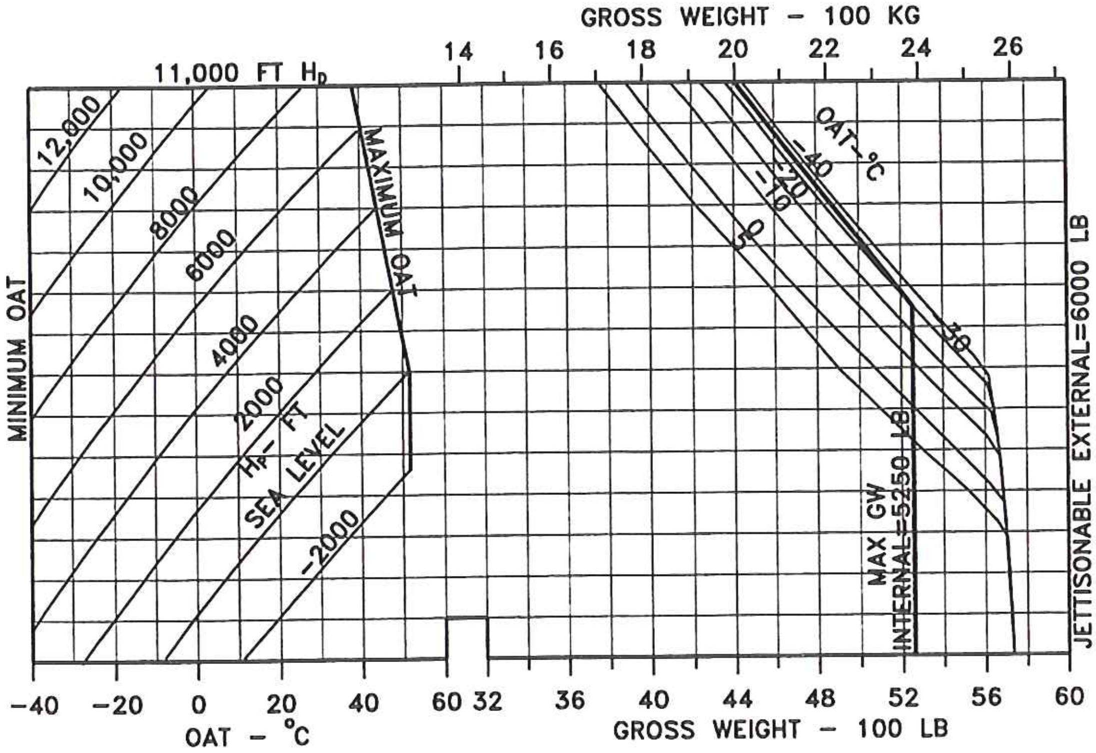

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Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 14 of 16)

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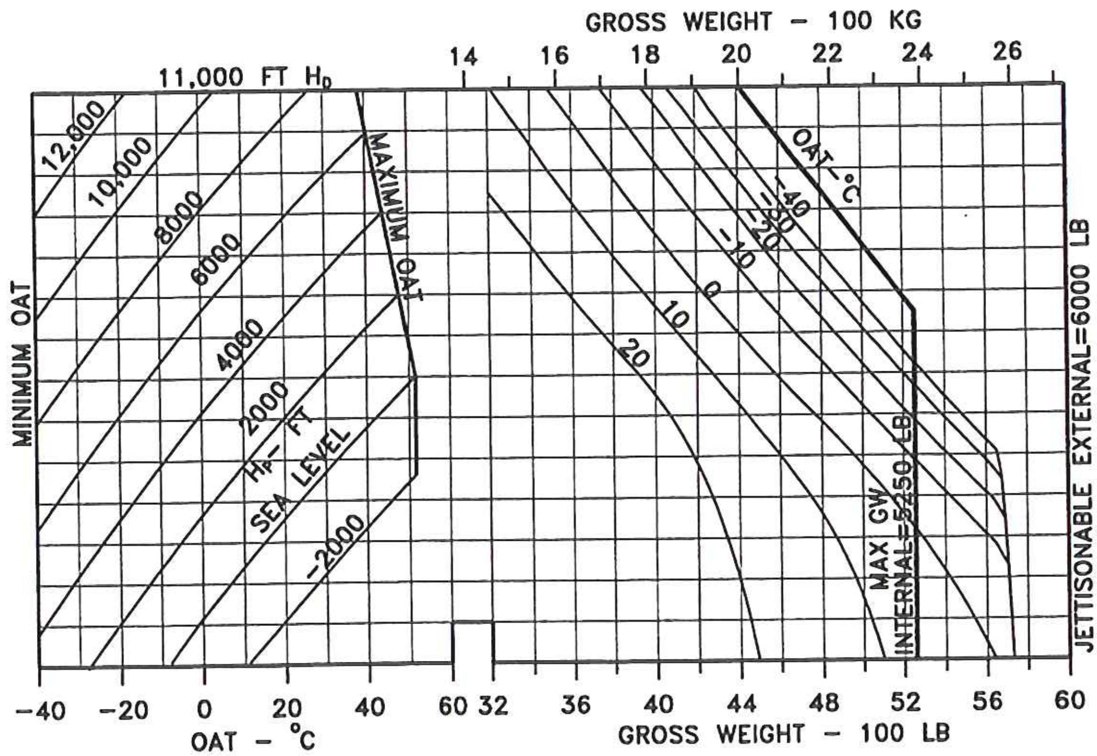
HOVER CEILING
OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER ON
PARTICLE SEPARATOR
SNOW BAFFLES



407FS28-4-6-15

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 15 of 16)

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HOVER CEILING

OUT OF GROUND EFFECT

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

SKID HEIGHT 40 FT (12.2 METER)
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR
SNOW BAFFLES

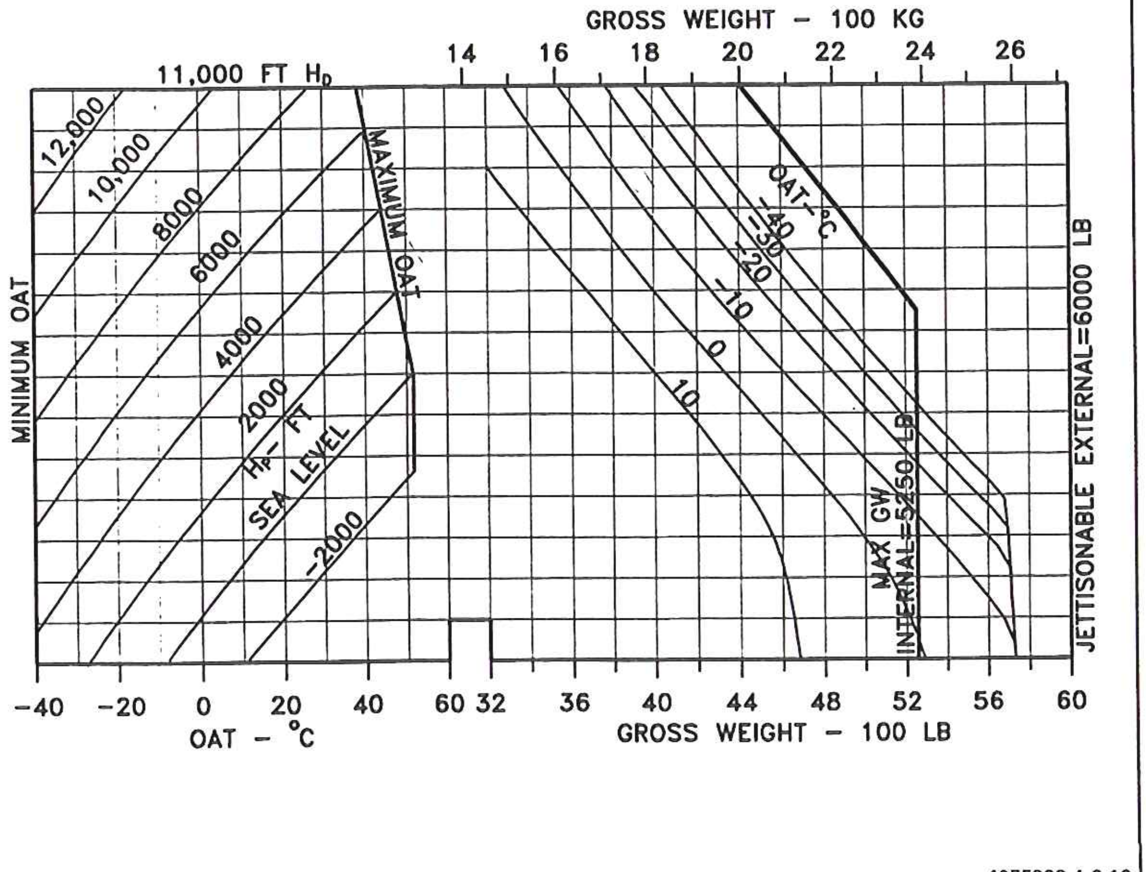

Figure 4-6. Hover ceiling OGE - maximum continuous power (Sheet 16 of 16)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER AND ANTI-ICE OFF
BASIC INLET

GROSS WEIGHT = 5250 LB

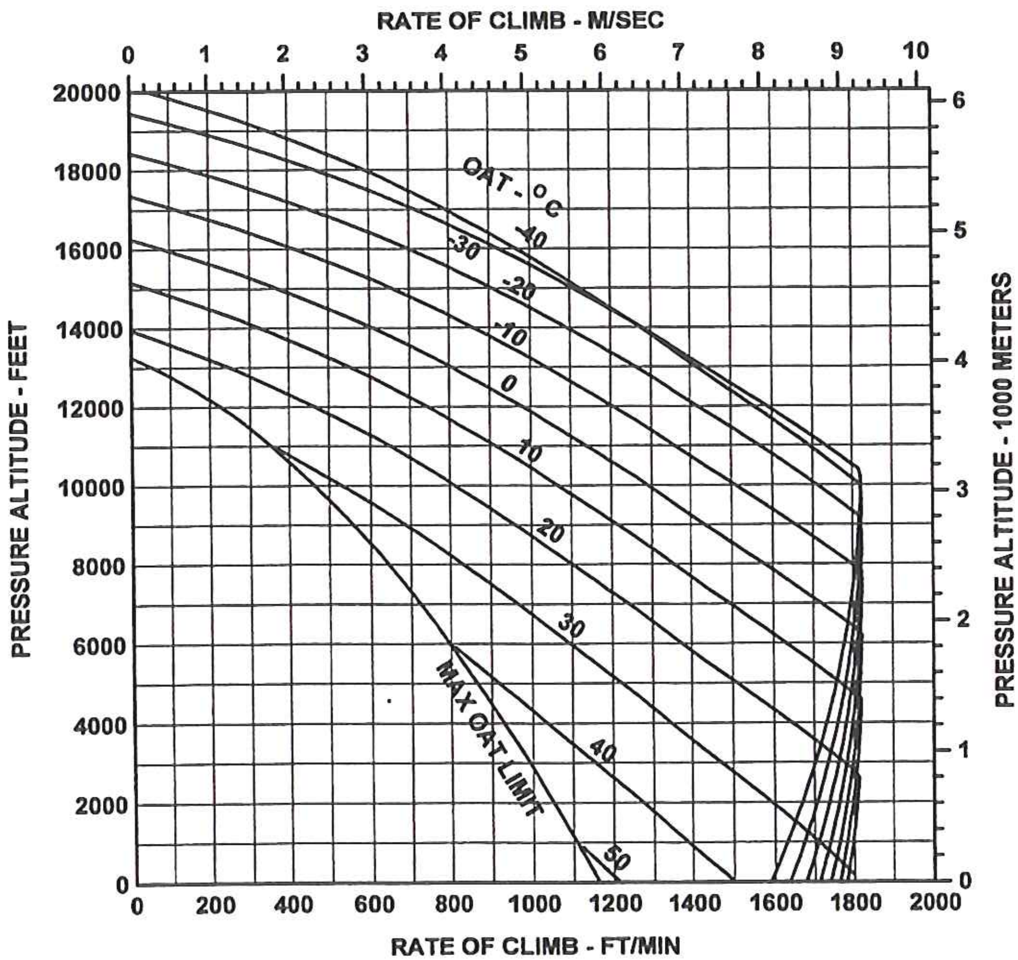

Figure 4-7. Rate of climb (takeoff power) (Sheet 1 of 4)

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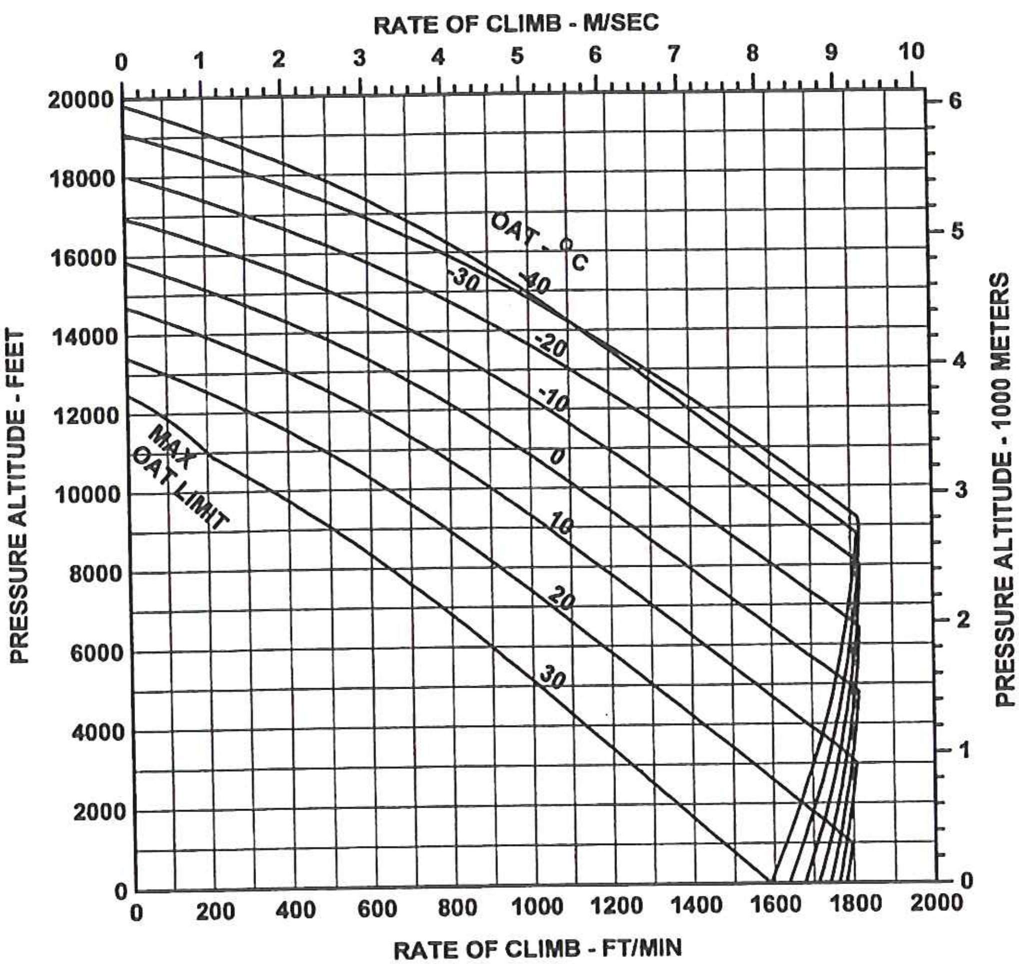
RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER AND ANTI-ICE ON
BASIC INLET
SNOW DEFLECTOR

GROSS WEIGHT = 5250 LB



407FS28-4-7-2

Figure 4-7. Rate of climb (takeoff power) (Sheet 2 of 4)

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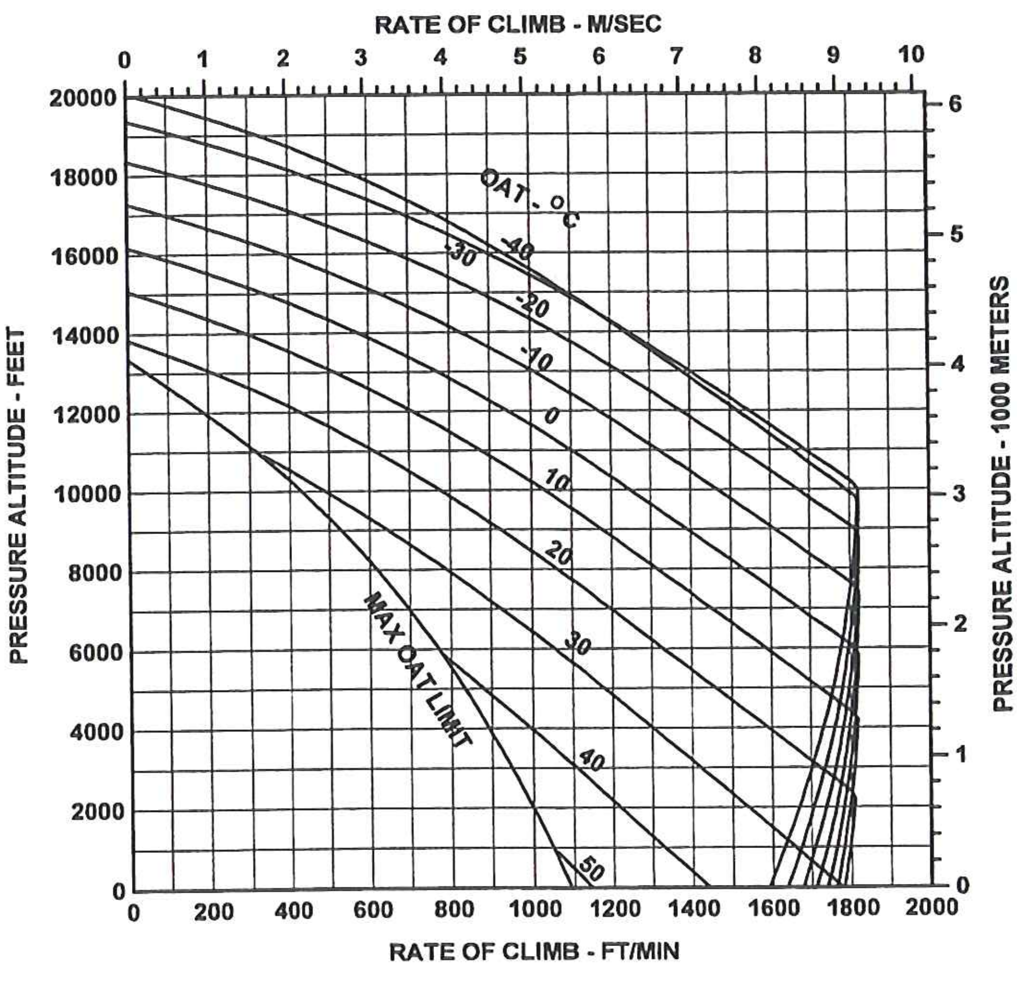
RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR

GROSS WEIGHT = 5250 LB



407FS28-4-7-3

Figure 4-7. Rate of climb (takeoff power) (Sheet 3 of 4)

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RATE OF CLIMB

TAKEOFF POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR
SNOW DEFLECTOR

GROSS WEIGHT = 5250 LB

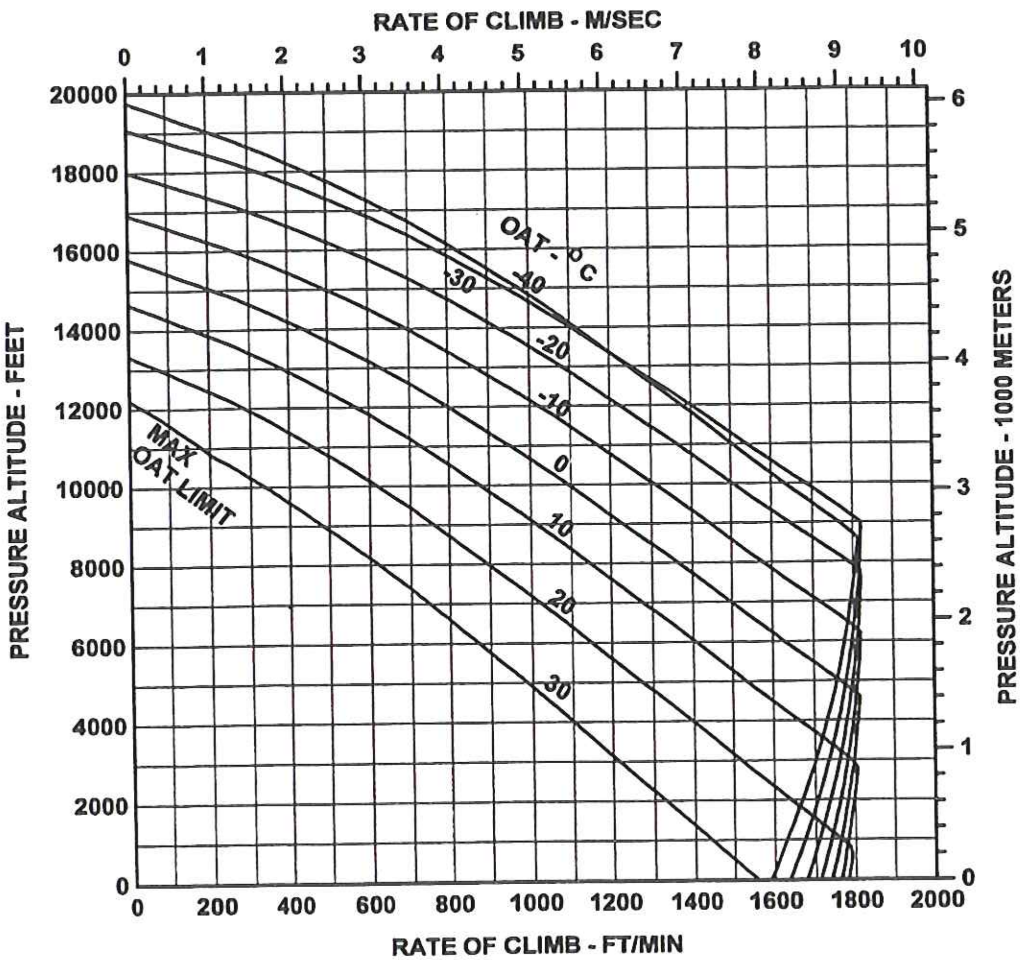

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Figure 4-7. Rate of climb (takeoff power) (Sheet 4 of 4)

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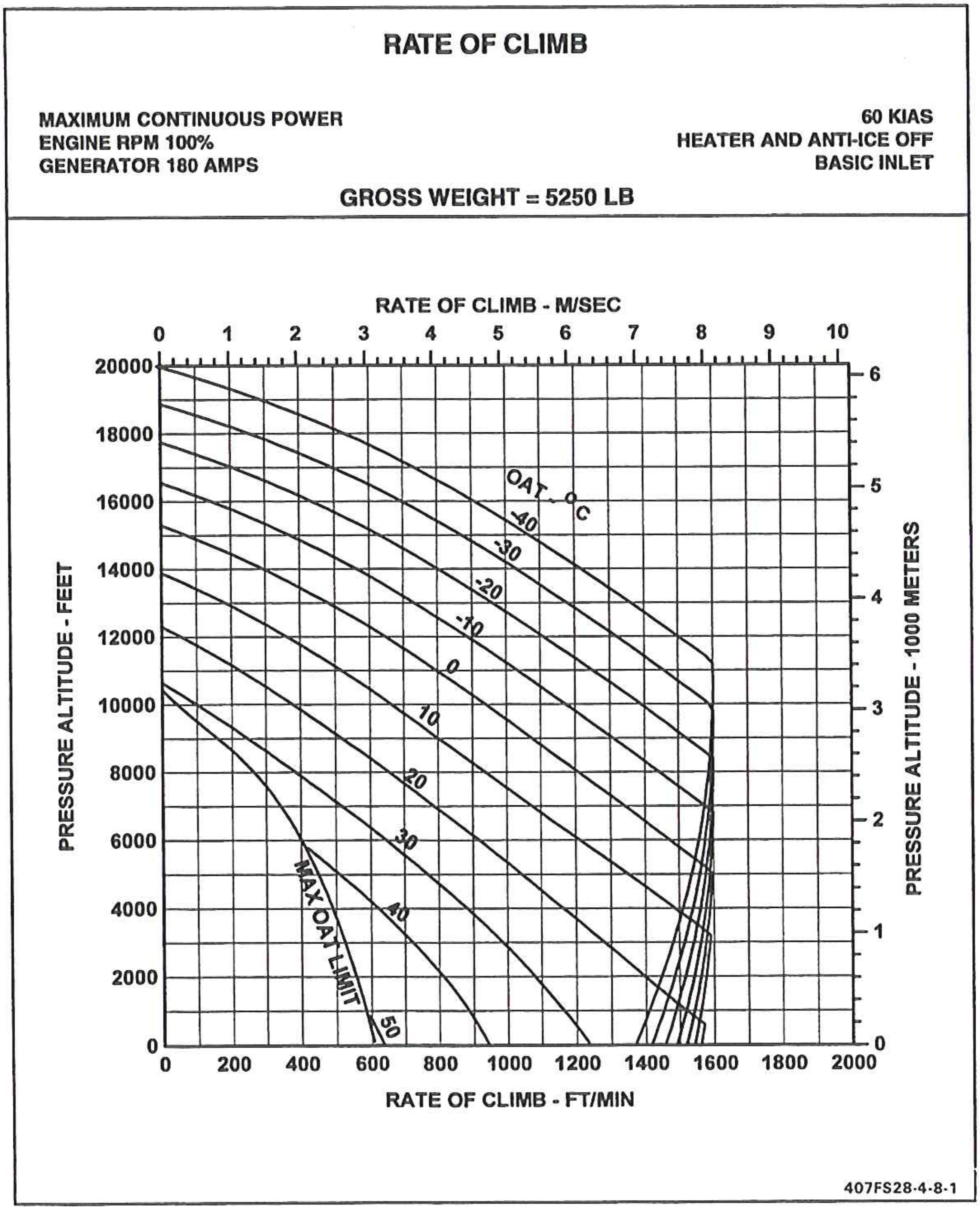

Figure 4-8. Rate of climb (maximum continuous power) (Sheet 1 of 4)

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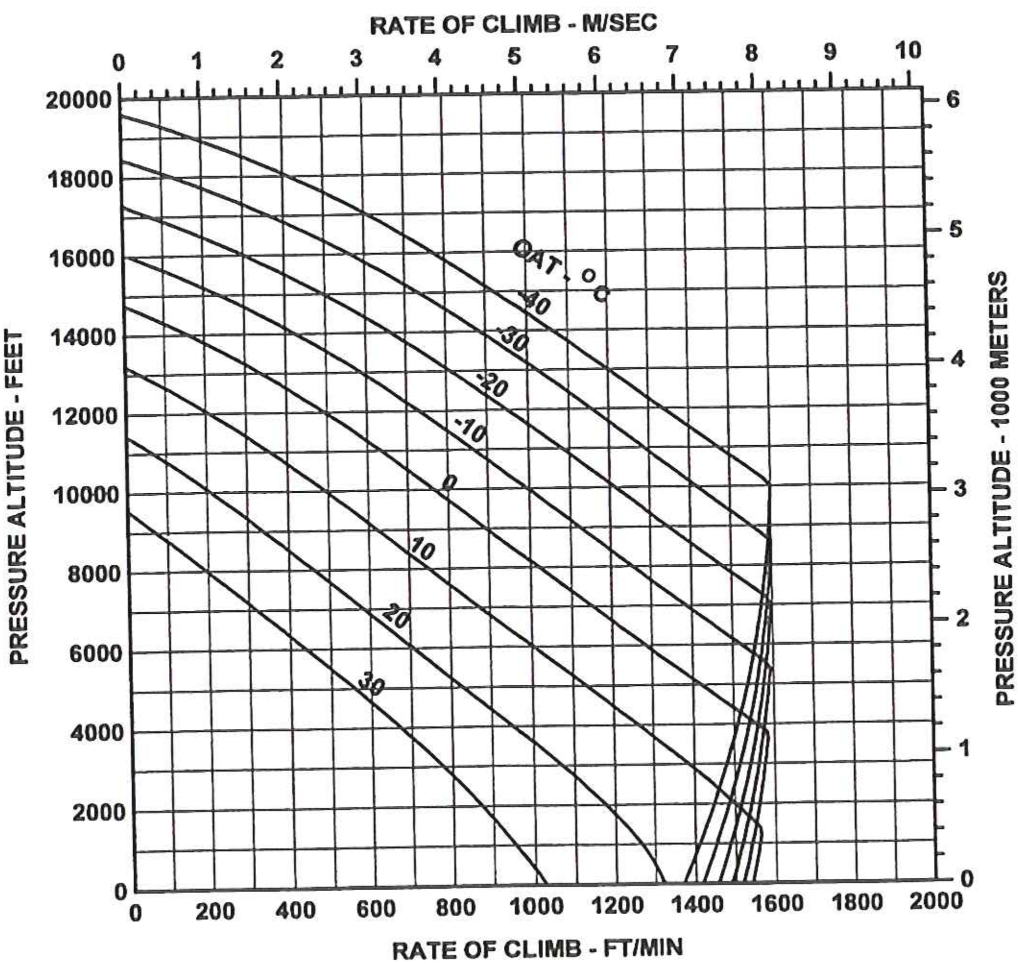
RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER AND ANTI-ICE ON
BASIC INLET
SNOW DEFLECTOR

GROSS WEIGHT = 5250 LB



407FS28-4-8-2

Figure 4-8. Rate of climb (maximum continous power) (Sheet 2 of 4)

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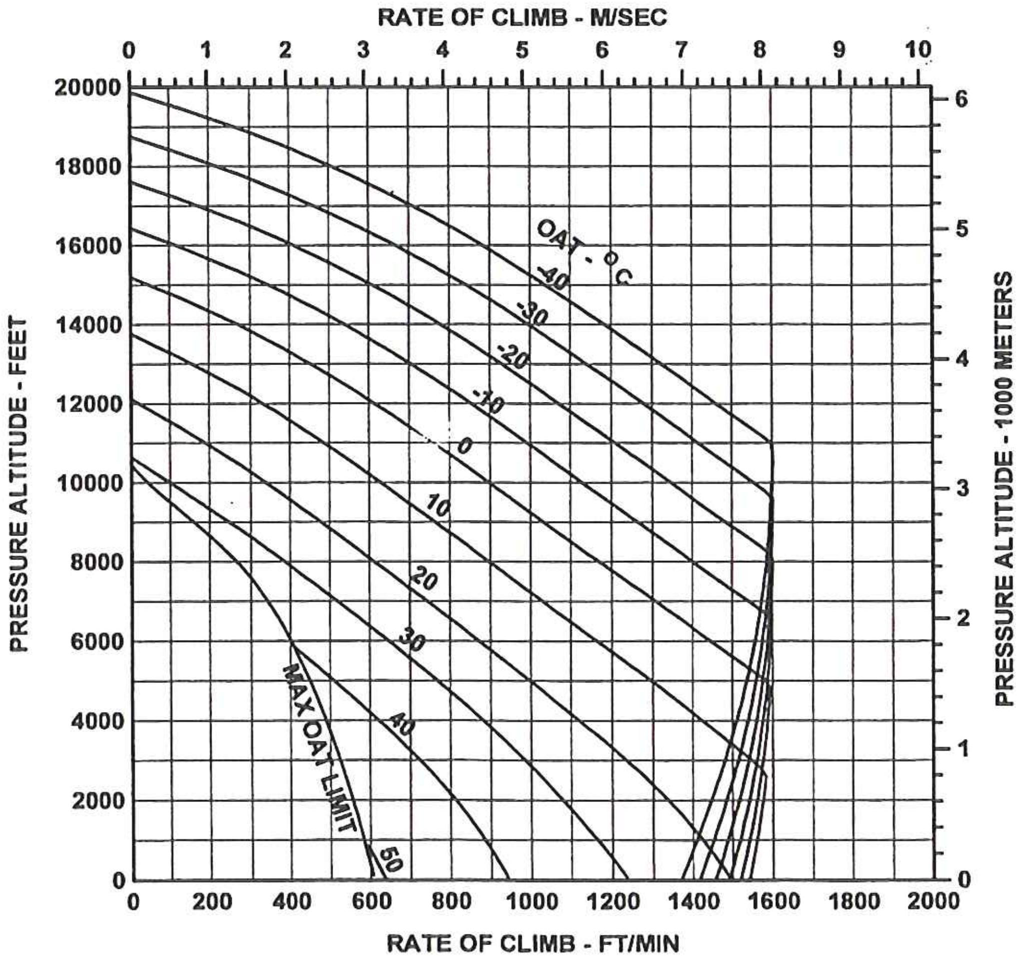
RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER AND ANTI-ICE OFF
PARTICLE SEPARATOR

GROSS WEIGHT = 5250 LB



407FS28-4-8-3

Figure 4-8. Rate of climb (maximum continuous power) (Sheet 3 of 4)

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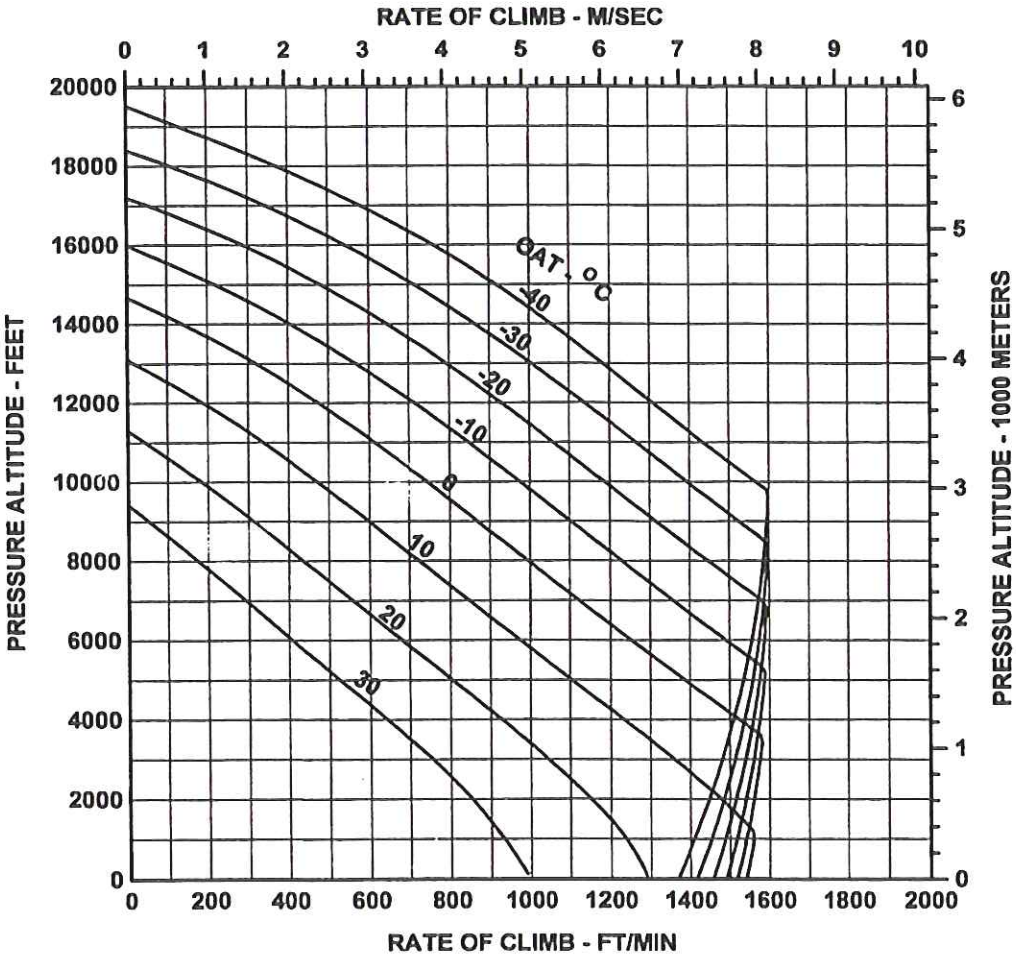
RATE OF CLIMB

MAXIMUM CONTINUOUS POWER
ENGINE RPM 100%
GENERATOR 180 AMPS

60 KIAS
HEATER AND ANTI-ICE ON
PARTICLE SEPARATOR
SNOW DEFLECTOR

GROSS WEIGHT = 5250 LB



407FS28-4-8-4

Figure 4-8. Rate of climb (maximum continuous power) (Sheet 4 of 4)

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1-23. ENGINE CONTROLS — FADEC SYSTEM

This paragraph addresses the operational design of the Rolls-Royce Allison 250-C47B FADEC system, its relationship with airframe and rotor systems, possible system faults, troubleshooting and training procedures. The information provided reflects operation with FADEC software version 5.202 and direct reversion to manual system installed. Although Flight Manual Emergency Procedures are explained in this section, refer to BHT-407-FM-1, Section 3, for actual flight operations.

1-23-A. FADEC SYSTEM

The 407 is in many areas a totally new design. This is particularly true in the area of the Full Authority Digital Electronic Control (FADEC) system. The FADEC includes numerous features and benefits never before available in a light, single-engine Bell helicopter. The FADEC system is designed to enhance flight safety and reduce pilot workload as well as provide other important benefits. In addition to the operational benefit of increased TBO, engine automatic start, and precise control of main rotor speed, are features such as redundant signal sensing, continuous monitoring and self diagnostics. Since much of the redundant design is transparent to the pilot, the caution / warning / advisory system has been expanded to advise of conditions resulting from the increased monitoring.

Although the possibility of a FADEC system failure is unlikely, Pilots and maintenance personnel must have an operational understanding of the FADEC system, along with a sound knowledge of emergency and troubleshooting procedures. Bell Helicopter recommends that personnel involved with the 407 familiarize themselves with the procedure for FADEC FAILURE. Familiarization with this procedure will help to emphasize Flight Manual Emergency Procedures as the following FADEC information is read within Manufacturers Data.

A FADEC FAILURE condition during operation in AUTO mode, will be indicated by activation of the FADEC FAIL warning horn and illumination of the FADEC FAIL and FADEC MANUAL warning lights.

Respond to the horn and lights as described in Section 3, Emergency/Malfunction Procedures, Basic Flight Manual.

1-23-B. FADEC SYSTEM - OPERATION

The control system (Figure 1-9) is designed to operate in many modes, covering all possible system states, including the transitional modes to and from the AUTO and MANUAL modes.

The FADEC uses a single channel control with 1 microprocessor and 1 electronic lane. There is also a MANUAL mode hydro mechanical back-up. The FADEC system has two main components. The airframe mounted Electronic Control Unit (ECU) and the engine mounted Hydro mechanical Unit (HMU).

The ECU monitors numerous internal and external inputs to modulate fuel flow and therefore control engine speed, acceleration rate, temperature and other engine parameters. The ECU provides inputs to the HMU to modulate fuel flow based on the continuous monitoring of the following: Measured Gas Temperature (MGT), Gas Producer speed (NG), Power Turbine speed (NP), Main Rotor speed (NR), Engine Torque Meter Oil Pressure (TMOP), Collective Pitch (CP) and rate, Compressor Inlet Temperature

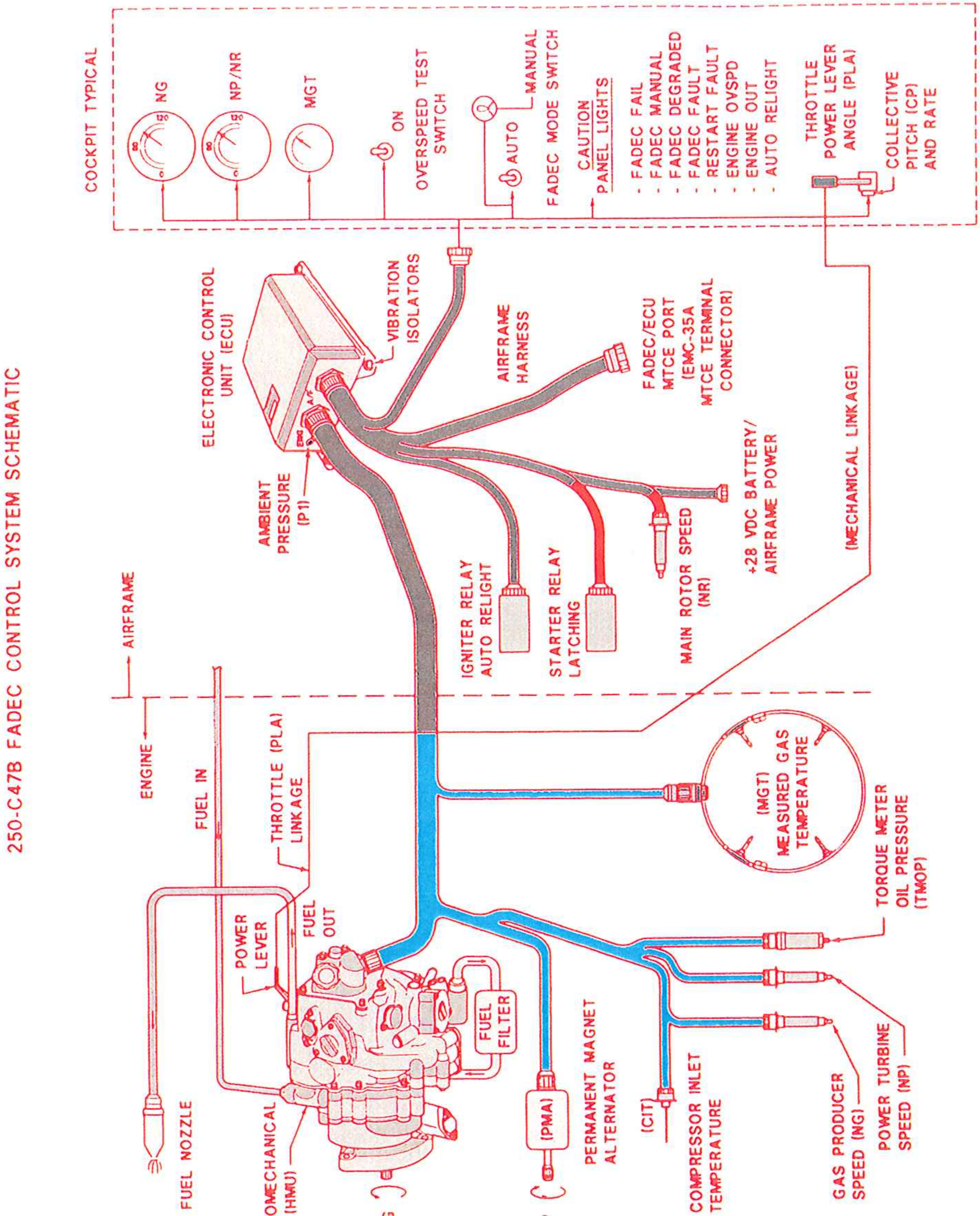

Figure 5-1 FADEC control system schematic

(CIT), Ambient Pressure (P1), and Power Lever Angle (PLA) / throttle position.

The HMU consists of a two stage suction fuel pump, fuel metering assembly, Auto/Manual changeover solenoid valve, electric overspeed valve, mechanical fuel shut off valve, hot start fuel solenoid valve and altitude compensated bellows. The HMU provides fuel modulation via a stepper motor in AUTO mode and a Hydro mechanical actuator in MANUAL mode.

1-23-C. POWER UP MODE AND BUILT IN TEST

The FADEC system incorporates logic and circuitry to perform self-diagnostics. In general, sensors are checked for continuity, rate and proper range. Discrete inputs are checked for continuity and output drivers are monitored for current demand to sense failed actuators and open or shorted circuits. A FADEC power up check exercises output drivers and actuators to ensure system functionality and readiness.

If any faults are detected during the self-test, the appropriate FADEC caution panel light will illuminate.

The helicopter 28 VDC bus supplies electrical power to the FADEC ECU until the engine achieves 85% NP. Above this speed, the FADEC ECU will select between the 28 VDC bus and the engine-driven permanent magnet alternator (PMA), as its primary power source. The higher voltage source will be selected. In the event of a primary power source failure, the alternate source will be selected.

1-23-D. START IN AUTO MODE

To ready the system for an automatic start, the FADEC MODE switch must be set to AUTO, and the throttle set to the idle position. The start switch is then momentarily positioned to START. Observe START and AUTORELIGHT lights are illuminated before releasing START switch. Throttle modulation of fuel flow is not required.

Although the start sequence is automatic, the pilot is responsible for monitoring the start and taking appropriate action if required. Therefore, it is recommended that both the throttle and start switch are guarded until the start is completed. Do not initiate a start if FADEC related caution panel lights are illuminated unless appropriate maintenance investigation or successful corrective action has been carried out and no "current" faults are shown.

NOTE

After the throttle is set to idle, the momentary contact start switch must be activated within 60 seconds to initiate the start and engage the latching feature. The latching feature of the start will engage when the FADEC ECU senses momentary activation (1 second) of the start switch or upon sensing an NG speed of 5%. If a start is attempted following a delay of more than 60 seconds, the FADEC system will not allow the starter to latch following release of the start switch, and will not introduce fuel if the start switch is held to START. Therefore, if a delay of more than 60 seconds has occurred, the system must be reset. To reset the system, the throttle must be repositioned to cutoff and then back to idle. In addition, if electrical power is interrupted prior to initiating the start, with the throttle at idle, the throttle must be repositioned to cutoff and then back to idle after power is restored to re-enable the latching feature. A normal automatic start sequence may then commence. If starting engine on external power, refer to paragraph 1-43A, External Power.

To allow for cooler starts and reduce the possibility of reaching hot start abort limits, it is recommended that residual MGT be below 150°C, when below 10,000 feet HP or below 65°C, when above 10,000 feet HP prior to start. To reduce residual MGT, a Dry Motoring Run may be performed in accordance with the Flight Manual.

Activating the automatic start mode engages the airframe mounted FADEC/start relay which is then latched by the FADEC ECU until the NG speed reaches 50%. The FADEC/ start relay places the MGT indicator into the start mode, signals the generator control unit / voltage regulator to inhibit generator output, flashes the shunt field for the duration of the start, and activates the starter relay. The starter relay activates the starter, illuminates the START advisory segment on the caution panel, and activates the igniter relay. The igniter relay activates the engine igniter system and illuminates the AUTO RELIGHT advisory segment of the caution panel. While in start mode, the MGT indicator is programmed to record start MGT exceedances, should they occur, which differ from normal operational MGT exceedance limits.

Once NG speed reaches 12%, for ambient temperatures greater than -6.7°C (20°F), or 10%, for ambient temperatures of -6.7°C (20°F) or less, the FADEC system will introduce fuel, detect the lightoff and smoothly accelerate the engine to idle while limiting MGT if necessary. At inlet temperatures below -18°C, the FADEC will increase start acceleration from 2% to 6% NG per second. The increase in the acceleration rate will be noticeable during start and improves start performance at colder temperatures and at higher altitudes.

Upon reaching an engine NG speed of 50%, the FADEC ECU unlatches the FADEC/start relay, terminating the start sequence. In addition, the ECU increments the start-counter to the next number when lightoff is detected.

Above 50% NG, the FADEC ECU carries out a self test of the auto relight system and continues to energize the igniter relay until 60±1% NG. During this time the AUTO RELIGHT light will be illuminated. The engine will continue to accelerate until reaching a stabilized idle of 63±1% NG.

The FADEC system also incorporates "Hot Start Abort Logic", up to 50% NG during start. This feature will cut off fuel flow to the engine fuel nozzle if any of the following conditions occur:

Start MGT exceeds 843°C, at pressure altitudes less than 10,000 feet and if residual MGT was less than 82.2°C at initiation (5% NG) of start.

Start MGT exceeds 912°C, at pressure altitudes greater than 10,000 feet or if residual MGT was greater than 82.2°C at initiation (5% NG) of start.

Voltage to FADEC ECU drops below 10.3 VDC. As a significant momentary voltage drop occurs at initiation of the start, ensuring a battery voltage of 24 VDC or above prior to start, in conjunction with appropriate battery maintenance, will reduce the possibility of voltage dropping to 10.3 VDC.

If a FADEC aborted start occurs or the pilot manually initiates a start abort by positioning the throttle to cut off, the FADEC is designed to automatically keep the starter engaged for up to 60 seconds from initiation of the start, to reduce MGT to 150°C. Once 150°C MGT is obtained, the starter will disengage.

NOTE

Momentarily positioning the start switch to DISENG will only deactivate the FADEC/start relay which in turn disables the starter and igniter circuits. In the event deactivation of the starter and igniter circuits occurs after engine light-off, but below 50% NG, the FADEC ECU will either modulate fuel flow to provide a start if NG speed is sufficient, or cutoff fuel flow if MGT exceeds start limits due to low NG speed. Therefore, positioning the throttle to cutoff is the appropriate method to manually stop the start sequence.

Following start and prior to increasing engine speed and main rotor rpm to 100%, a FADEC MANUAL check is required. The purpose of this check is to ensure the engine responds to throttle movement while in MANUAL mode. Prior to positioning the FADEC mode switch to MANUAL for the purposes of this check, ensure the throttle is positioned to idle and NG speed is at 63± 1%.

Once the FADEC mode switch is positioned to MANUAL, the FADEC MANUAL and AUTO RELIGHT lights will illuminate. In addition, the NG speed may change from 63± 1%. An increase or decrease in NG speed may be noticed. The change in NG speed is due to the fact that the FADEC system is not temperature compensated in regard to fuel flow in MANUAL mode. Engine load and HMU calibration will also play a part in determining if the NG speed will change. If a change in NG speed is noted adjust the throttle to maintain an NG idle speed of 63± 1%.

Once stabilized at an idle speed of 63± 1% NG, slowly increase throttle to ensure engine responds and then return to idle speed. Position FADEC Mode switch to AUTO and ensure FADEC MANUAL and AUTO RELIGHT lights extinguish.

Engine acceleration from idle to 100% NP / NR, in AUTO mode, is achieved by smoothly increasing the throttle to the FLY detent position (PLA 70°). As the throttle is positioned from idle (PLA 30° to 40°) to the FLY detent position (PLA 70°), electrical signals are sent to the ECU from the HMU - PLA potentiometer. These signals dictate the amount of authority the ECU has to control maximum fuel flow (NG limiting) based on throttle position, and in turn controls engine NG speed. Therefore, as the throttle is increased from idle to the FLY detent position, the fuel flow is electronically increased until 100% NP/NR is obtained. To avoid rapid engine acceleration, it is recommended that throttle application from idle to the FLY detent position be conducted in a smooth and gradual manner.

1-23-E. START IN MANUAL MODE

In accordance with the Rolls-Royce Allison 250-C47B Operation and Maintenance Manual, Manual Mode starting on the ground is not authorized except for use under emergency conditions or under special permit from the local aviation authority. Refer to the Rolls Royce Allison 250-C47B Operation and Maintenance Manual to ensure all Manual Mode Operational Procedures are followed. Automatic hot start abort features are not available in MANUAL mode.

To ready the system for a MANUAL start, the FADEC MODE switch is set to MAN and the throttle positioned to cutoff. In MANUAL, the FADEC will not latch the FADEC / start relay or control the fuel scheduling during the start. The start switch must be held in the START position until the start sequence is completed at 50% NG. When the NG speed reaches 12%, or NG specified in Allison 250-C47B operation and maintenance manual for specific OAT, slowly advance the throttle out of cutoff and stop when the engine lights off. Allow the MGT to peak and then increase fuel flow by modulating throttle to maintain MGT within limits. Idle speed in MANUAL may not stabilize at 63±1% NG. If this occurs, maintain idle speed at 63±1% NG with throttle.

Once a MANUAL mode start has been initiated, it may be terminated at any time by rotating the throttle to cutoff. The pilot should continue motoring the engine until the MGT has stabilized at an acceptable level. Releasing the start switch from the START position will disengage the FADEC / start relay and disable the starter and igniter circuits.

After a successful MANUAL mode start, and idle has been achieved and is stable, perform a momentary switch back to AUTO mode. If engine flameout occurs, subsequent starts/flight are prohibited until appropriate FADEC system troubleshooting has been performed.

Engine acceleration from idle to 100% is achieved by positioning the throttle towards full open. This is achieved through hydro mechanical control of the HMU fuel metering valve.

1-23-F. IN-FLIGHT - AUTO MODE OPERATION

NOTE

If throttle is not maintained in FLY detent position during normal flight operations, available engine power may be limited. This will occur if throttle is positioned from the FLY detent (PLA 70°) to a setting which is less than 62° PLA. Although any throttle position from 62° PLA to full open will provide the FADEC with complete control to maintain NR within limits; the approved throttle position for flight is the FLY detent position (PLA 70°).

During flight in AUTO mode with the throttle in FLY detent position (PLA 70°), the FADEC has complete control over engine operation to maintain NR within limits. The ECU receives engine and airframe parameter inputs and cockpit command signals, processes them and modulates the HMU stepper motor driven fuel metering valve to achieve desired engine performance.

If required, as may be the case in certain Emergency Procedures, an alternate means of engine control is also available to the pilot. This can be achieved by manipulating the throttle below FLY detent position until the required engine performance is achieved. As the throttle is positioned between HMU Power Lever Angles of 40° to 62°, electrical signals are sent to the ECU from the HMU-PLA potentiometer. These signals dictate the amount of authority the ECU has to control maximum fuel flow (NG limiting), and in turn, engine NG speed. Therefore, as throttle is increased or decreased, the maximum NG speed is regulated electrically by limiting the fuel flow.

In AUTO mode, the FADEC is capable of detecting an engine flameout by sensing NG deceleration. Without any pilot action, the auto-relight sequence is initiated by establishing a controlled fuel flow and activating the ignition system. The FADEC will control the MGT and accelerate the engine to the previously selected state.

The automatic relight sequence will initiate at detection of a flameout, continuing the procedure until a relight occurs or the NG speed decays to 50%. During this time the AUTO RELIGHT and ENGINE OUT caution panel lights will be illuminated and the engine out warning horn will be activated. If the NG decays below 50%, the FADEC system will discontinue the attempt to relight the engine and the ENGINE OUT light and warning horn will remain active. In the event of an unsuccessful relight, refer to the restart - automatic mode emergency procedure described in the Flight Manual. For additional information on in-flight AUTO mode restarts, refer to paragraph 1-29.

While in AUTO mode, if any failure occurs in the ECU or in one of its input/output signals that significantly impacts the ECU control of the HMU, the pilot will be alerted via the FADEC FAIL warning horn and the FADEC FAIL and FADEC MANUAL warning lights. If the detected failure does not significantly impair the functioning of the ECU, the pilot will be alerted via a FADEC DEGRADED, FADEC FAULT caution light, RESTART FUALT advisory light, or combination of, depending on the nature of the fault.

1-23-G. FADEC SYSTEM FAULTS

CAUTION

BELL HELICOPTER REQUIRES MAINTENANCE ACTION, PRIOR TO FLIGHT, WHEN A FADEC RELATED LIGHT IS ILLUMINATED.

There are eight lights in the caution/warning/advisory panel that are controlled by the FADEC: FADEC FAIL, FADEC MANUAL, FADEC DEGRADED, FADEC FAULT, RESTART FAULT, ENGINE OVSPD, ENGINE OUT, and AUTO RELIGHT.

The FADEC ECU continuously monitors the FADEC system for faults and makes appropriate accommodations to continue operation. Fault codes have been preassigned to those parameters being monitored by the FADEC ECU.

Faults and Exceedances can be recorded under the following conditions:

Engine operating:

When the engine is operating (i.e. lightoff has been detected) the FADEC will automatically record faults / exceedances as current, last engine run, and accumulated, as they occur.

Engine not operating:

When the engine is not operating, but electrical power is applied, the FADEC will only record current faults as they occur.

If any failure occurs in the ECU / HMU or in one of the input / output signals that significantly impacts the ECU or control of the HMU, the pilot will be alerted via the FADEC FAIL warning horn and the FADEC FAIL / FADEC MANUAL warning lights. If the detected failure does not significantly impair the functioning of the ECU, the pilot will be alerted via a FADEC DEGRADED, FADEC FAULT caution light, RESTART FAULT advisory light, or a combination of, depending upon the nature of the fault.

If the fault is minor in nature, it will not be communicated to the pilot with the engine running. These faults are identified as maintenance advisory faults and will be displayed during shutdown when the throttle is placed in the cutoff position and NG speed decays below 9.5%. This will be in the form of a FADEC DEGRADED light.

The Basic Flight Manual provides the appropriate action required by the pilot for each light or light / horn condition.

All FADEC faults have been categorized into five types. The first four relate to in-flight faults and the fifth relates to Maintenance Advisory faults with the engine shut down. Maintenance Advisory faults displayed during shutdown will be discussed under the heading FADEC SYSTEM FAULTS - ENGINE SHUT DOWN.

1-23-H. CATEGORY 1 - FADEC FAIL / FADEC MANUAL — DIRECT REVERSION TO MANUAL SYSTEM (DRTM)

Faults that require pilot action and automatically initiate a transition to the MANUAL mode will be displayed immediately when detected by the ECU. These faults will display as FADEC FAIL / FADEC MANUAL. The FADEC FAIL horn (chime tone) will activate in conjunction with the FADEC FAIL and FADEC MANUAL warning lights. In addition, the RESTART FAULT light will also be displayed with a FADEC FAIL / FADEC MANUAL condition.

The DIRECT REVERSION to MANUAL system ensures all FADEC failures revert directly to MANUAL. FAIL FIXED failures do not exist with this system. In addition, the system incorporates a throttle which is detented at the 90% NG bezel FLY position.

The main intent of DIRECT to MANUAL system is to simplify pilot procedures in the event of a FADEC failure. This is accomplished by allowing the pilot to keep his hands on the controls during a FADEC failure and enable an increase or decrease in throttle from the FLY detent position as required. The pilot will only have to remove his hand from the collective to press the FADEC MODE switch to silence the horn once established in manual mode.

1-23-H-1. THE FADEC SYSTEM WILL FAIL TO THE MANUAL MODE AS FOLLOWS:

In this situation, reversion to MANUAL mode will occur independent of the position of the FADEC MODE switch on the instrument panel. The reversion to the MANUAL mode will begin immediately.

This condition can be identified by sounding of the FADEC FAIL warning horn, illumination of both the FADEC FAIL and FADEC MANUAL caution panel lights and FADEC MODE switch MAN light at the time of the failure. As the FADEC SYSTEM has initiated the transition to MANUAL mode, the pilot must be aware that an increase or decrease in NP/NR may occur within 2 to 7 seconds following a direct failure to MANUAL. If this occurs, collective and throttle will have to be used to control RPM.

The objective of the DIRECT REVERSION to MANUAL system is to provide the pilot with throttle control of the fuel metering valve in a timely manner. MANUAL mode allows the pilot to control NP/NR with coordinated control of the collective and throttle.

The following procedural steps are required:

1. Throttle — If time permits, match throttle bezel position to NG indication.

    This has proven to reduce the possibility of an overspeed or underspeed condition.

    This procedure also permits a smoother transition to MANUAL. This is due to the fact that the actual NG speed and fuel metering valve position prior to switching to MANUAL will be very close to that following the transition to MANUAL mode, resulting in little, if any, RPM change.

    If time does not permit matching of throttle bezel position to NG indication, initially maintain throttle in detented FLY position.

2. NR / NP — Maintain 95 to 100% with collective and throttle.

It is most important to ensure that NR/NP is monitored and properly controlled, during and following the transition to MANUAL.

Within 2 to 7 seconds after FADEC FAIL warning, NR /NP may increase or decrease very rapidly. This will require collective inputs to control RPM. The 407 rotor system is very responsive to collective inputs and can be controlled by the pilot should a NP/NR overspeed/ underspeed tendency arise.

As it takes 2 to 7 seconds to complete the transition to MANUAL mode, use of throttle to control NR /NP will be ineffective until the transition to MANUAL mode is complete. The transition will not be completed until the fuel metering valve in the HMU can be manually controlled by the pilot through use of the throttle on the collective.

There are two pistons within the HMU (Figure 1-10), a Manual Load Piston (slow piston) and a PLA Follower Piston (fast piston), which must hydro mechanically extend to contact opposite sides of the fuel metering valve shaft lever. The two pistons move at different rates toward the fuel metering valve lever. It takes approximately 2.0 seconds for both pistons to make contact with the fuel metering valve lever following a transition from an initial condition of low fuel flow. Similarly, up to 7 seconds may be required for the two pistons to make contact following a transition from an initial condition of high fuel flow. Refer to Figures 1-10, 1-11 and 1-12 for additional information on AUTO to MANUAL transitions.

An increase in NP/NR speed may be experienced while in transition to MANUAL from a condition of low to higher fuel flow or high fuel flow to a higher fuel flow. This will be seen if the throttle to bezel selection made by the pilot in Step 1 of the procedure is higher than the actual NG speed at the time of the FADEC FAILURE condition.

This will occur during the period when the HMU Manual Load Piston (slow piston) engages the fuel metering valve lever and moves it to a more open position until the PLA Follower Piston (fast piston) is contacted.

Inversely, a decrease in NP/NR speed may be experienced during the transition to MANUAL from a condition of low to lower fuel flow or high fuel flow to a lower fuel flow. This will be seen if the throttle to bezel selection made by the pilot in step 1 of the procedure is lower than the actual NG speed at the time of the FADEC FAILURE condition. This will occur during the period when the PLA Follower Piston (fast piston) engages the fuel metering valve lever and moves it to a more closed position as dictated by throttle to bezel position.

The approximate time to detect a power change during the transition to manual is summarized in Table 1-1.

In simpler terms, there will be a time delay and possible change in engine power while the system transitions to MANUAL. The length of the delay and degree of power change during the transition depends on engine power at the time of the FADEC failure and the desired power as selected by throttle position for the transition. As stated previously, the degree of power change can be minimized by matching the throttle bezel to the actual indicated NG speed in step one of the FADEC FAILURE procedure. This permits the smoother transition to MANUAL due to the fact that the actual NG speed and fuel metering valve position prior to switching to MANUAL will be very close to that following the transition to MANUAL mode. This will result in little if any, power/RPM change.

Once both pistons contact the lever on the fuel metering valve shaft, the transition to MANUAL Mode will be complete. The pilot will have slew rate limited control of the fuel metering valve via throttle position without any delay.

Throttle may now be used, in conjunction with collective, to maintain rotor and engine RPM within 95 - 100%.

Table 1-1. Time to power change

ENGINE POWER AT TIME OF FADEC FAILURE	DESIRED POWER AS SELECTED BY THROTTLE POSITION	APPROX. TIME TO DETECT POWER CHANGE DURING TRANSITION TO MANUAL
LOW POWER	HIGHER POWER	2.0 SECONDS
LOW POWER	LOWER POWER	1.0 SECOND
HIGH POWER	HIGHER POWER	7.0 SECONDS
HIGH POWER	LOWER POWER	0.1 SECOND

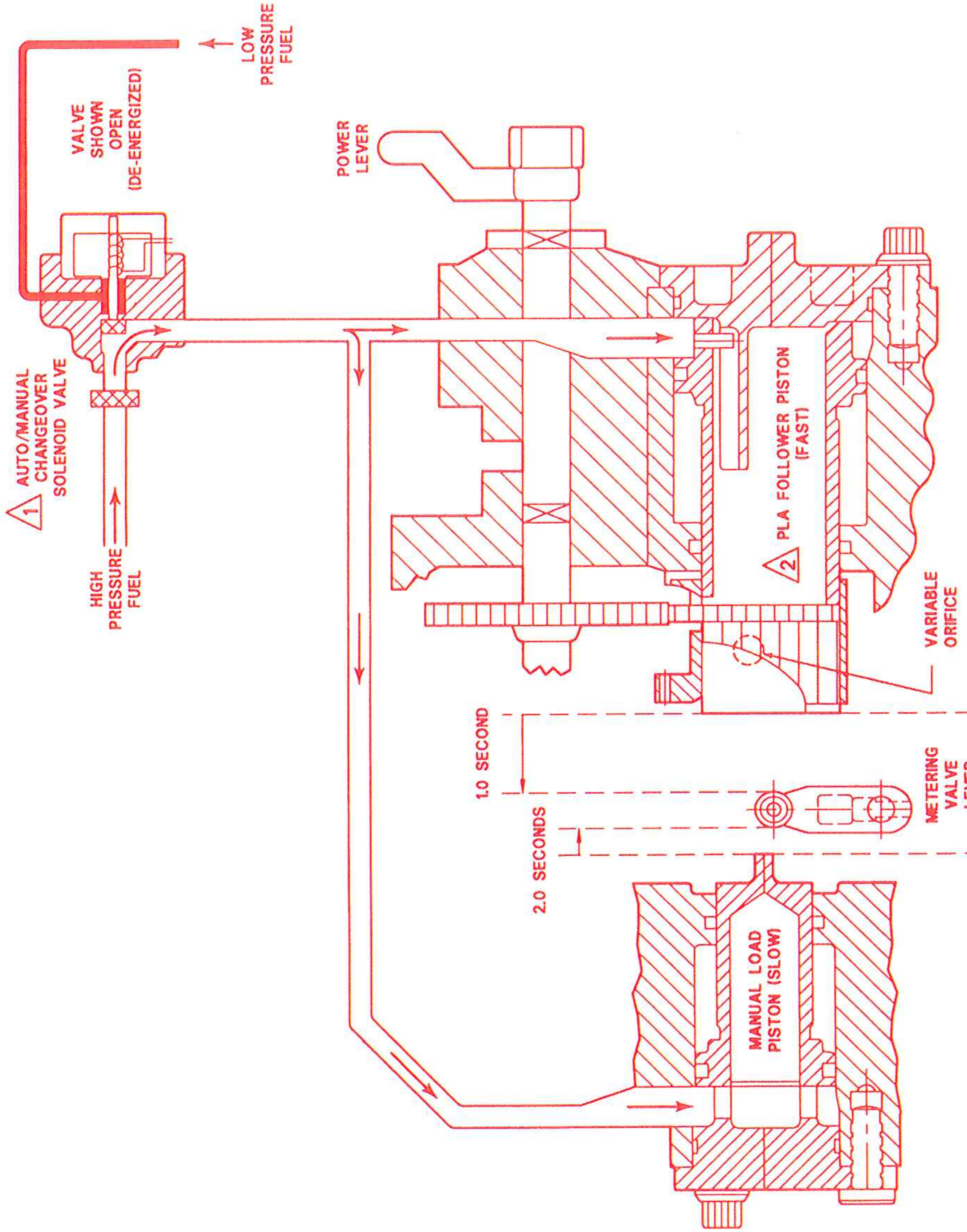

Figure 5-2 Low fuel flow (Low power)

AUTO TO MANUAL TRANSITION AT LOW FUEL FLOW

INITIAL CONDITION:

- Auto mode, low engine fuel flow, throttle FLY detent position.

After FADEC mode switch manual selection or initiation of direct reversion to manual:

- Auto/manual changeover solenoid valve is de energized allowing high pressure fuel to manual loa piston and PLA follower piston.
- Manual load piston slowly extends. Engages metering valve lever in approximately 2.0 seconds an begins to drive it to meet PLA follower piston.
- Concurrently, PLA follower piston rapidly extends in approximately 1.0 second to a position that is function of throttle (PLA) position.
- Matching throttle and bezel to the actual NG speed will allow the PLA follower piston to position itself very close to the actual position of the metering valve at the time of the transition. This will minimize the fuel flow change during the transition. Positioning the throttle to a bezel setting that i higher than actual NG speed at the time of the transition will produce an increase in fuel flow durin the transition. The increase in fuel flow will be caused as the manual load piston engages the metering valve lever and drives it toward the PLA follower piston. Inversely, positioning the throttle to a bezel setting that is lower than actual NG speed at the time of the transition will produce a decrease in fuel flow during the transition. The decrease in fuel flow will be caused as the PLA follower piston engages the metering valve lever and drives it towards the manual load position.
- After both pistons engage, manual mode is established and no delay exists between throttle (PLA movement and fuel flow change.
- Slew rate limiting is achieved by hydraulic dynamics.

NOTES

1 - Auto/manual changeover solenoid valve normally closed (energized) in auto mode.

- Auto/manual changeover solenoid valve is opened (de-energized) for transition to and durinc manual mode operation. With the valve open, fuel pressure is used to position the manual loa piston and PLA follower piston.

2 - PLA follower piston is controlled by throttle (PLA) position during transition to manual and when in manual mode. PLA follower piston position is regulated by fuel pressure bleed through variable orifice. This provides the means to increase or decrease fuel flow by alterin the position of the fuel metering valve. Manual load piston ensures metering valve lever is hel against PLA follower piston.

- When in automatic mode, both the PLA follower piston and manual load piston are retracted from the metering valve lever. They are held in the retracted position by fuel pressure when th auto/manual changeover solenoid valve is closed (energized).

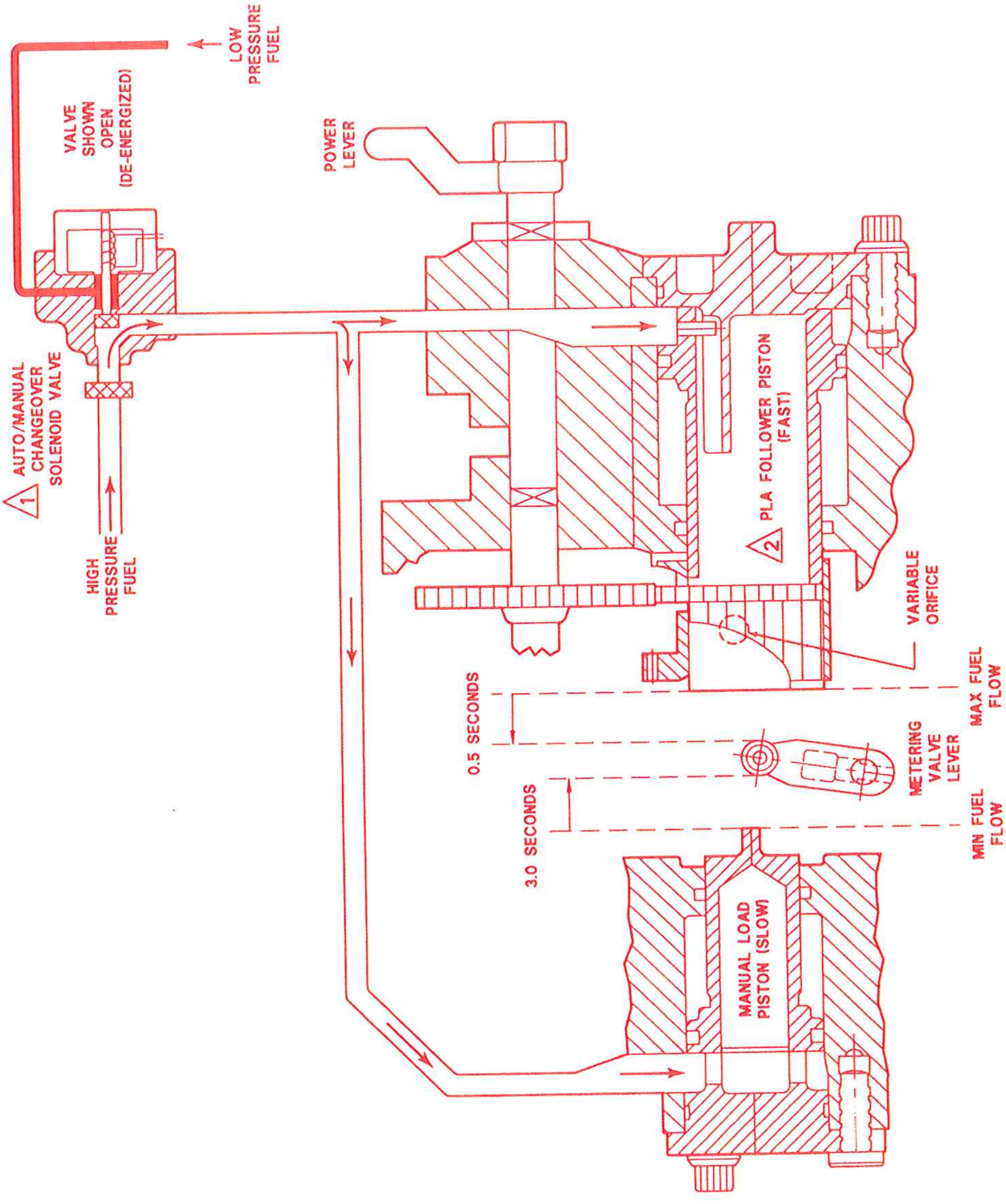

Figure 5-3 Intermediate fuel flow (Moderate power)

AUTO TO MANUAL TRANSITION AT INTERMEDIATE (CRUISE) FUEL FLOW

INITIAL CONDITION:

- Auto mode, intermediate engine fuel flow, throttle FLY detent position.

After FADEC mode switch manual selection or initiation of direct reversion to manual:

- Auto/manual changeover solenoid valve is de energized allowing high pressure fuel to manual load piston and PLA follower piston.
- Manual load piston slowly extends. Engages metering valve lever in approximately 3.0 seconds and begins to drive it to meet PLA follower piston.
- Concurrently, PLA follower piston rapidly extends in approximately 0.5 second to a position that is a function of throttle (PLA) position.
- Matching throttle and bezel to the actual NG speed will allow the PLA follower piston to position itself very close to the actual position of the metering valve at the time of the transition. This will minimize the fuel flow change during the transition. Positioning the throttle to a bezel setting that is higher than actual NG speed at the time of the transition will produce an increase in fuel flow during the transition. The increase in fuel flow will be caused as the manual load piston engages the metering valve lever and drives it toward the PLA follower piston. Inversely, positioning the throttle to a bezel setting that is lower than actual NG speed at the time of the transition will produce a decrease in fuel flow during the transition. The decrease in fuel flow will be caused as the PLA follower piston engages the metering valve lever and drives it towards the manual load position.
- After both pistons engage, manual mode is established and no delay exists between throttle (PLA) movement and fuel flow change.
- Slew rate limiting is achieved by hydraulic dynamics.

NOTES

△1 - Auto/manual changeover solenoid valve normally closed (energized) in auto mode.

- Auto/manual changeover solenoid valve is opened (de-energized) for transition to and during manual mode operation. With the valve open, fuel pressure is used to position the manual load piston and PLA follower piston.

△2 - PLA follower piston is controlled by throttle (PLA) position during transition to manual and when in manual mode. PLA follower piston position is regulated by fuel pressure bleed through variable orifice. This provides the means to increase or decrease fuel flow by altering the position of the fuel metering valve. Manual load piston ensures metering valve lever is held against PLA follower piston.

- When in automatic mode, both the PLA follower piston and manual load piston are retracted from the metering valve lever. They are held in the retracted position by fuel pressure when the auto/manual changeover solenoid valve is closed (energized).

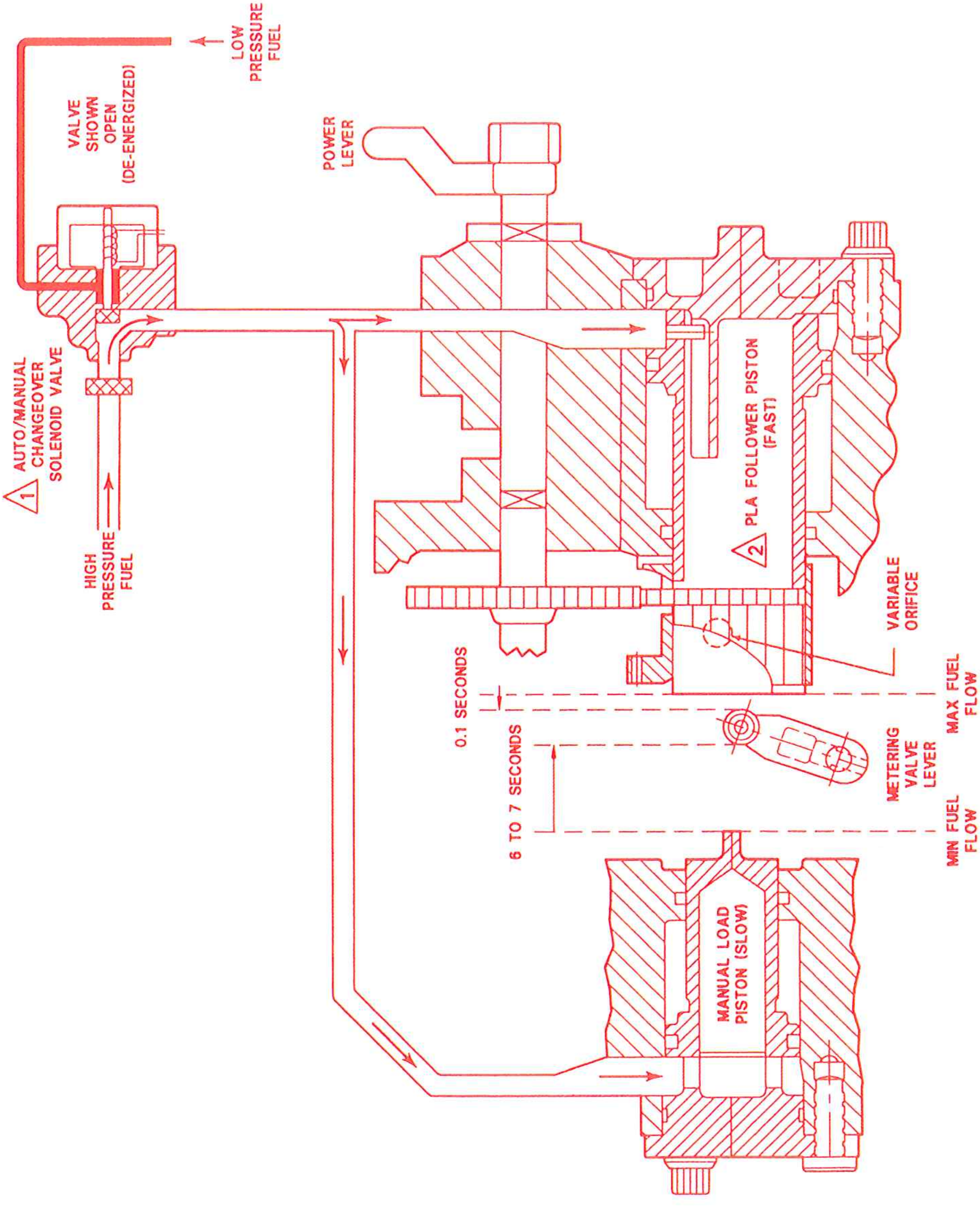

Figure 5-4 High fuel flow (High power)

AUTO TO MANUAL TRANSITION AT HIGH FUEL FLOW

INITIAL CONDITION:

- Auto mode, high engine fuel flow, throttle FLY detent position.

After FADEC mode switch manual selection or initiation of direct reversion to manual:

- Auto/manual changeover solenoid valve is de energized allowing high pressure fuel to manual loa piston and PLA follower piston.
- Manual load piston slowly extends. Engages metering valve lever in approximately 6 to 7 seconds and begins to drive it to meet PLA follower piston.
- Concurrently, PLA follower piston rapidly extends in approximately 0.1 second to a position that is function of throttle (PLA) position.
- Matching throttle and bezel to the actual NG speed will allow the PLA follower piston to position itself very close to the actual position of the metering valve at the time of the transition. This will minimize the fuel flow change during the transition. Positioning the throttle to a bezel setting that i higher than actual NG speed at the time of the transition will produce an increase in fuel flow durin the transition. The increase in fuel flow will be caused as the manual load piston engages the metering valve lever and drives it toward the PLA follower piston. Inversely, positioning the throttle to a bezel setting that is lower than actual NG speed at the time of the transition will produce a decrease in fuel flow during the transition. The decrease in fuel flow will be caused as the PLA follower piston engages the metering valve lever and drives it towards the manual load position.
- After both pistons engage, manual mode is established and no delay exists between throttle (PLA) movement and fuel flow change.
- Slew rate limiting is achieved by hydraulic dynamics.

NOTES

1 - Auto/manual changeover solenoid valve normally closed (energized) in auto mode.

- Auto/manual changeover solenoid valve is opened (de-energized) for transition to and during manual mode operation. With the valve open, fuel pressure is used to position the manual loa piston and PLA follower piston.

2 - PLA follower piston is controlled by throttle (PLA) position during transition to manual and when in manual mode. PLA follower piston position is regulated by fuel pressure bleed through variable orifice. This provides the means to increase or decrease fuel flow by alterin the position of the fuel metering valve. Manual load piston ensures metering valve lever is hel against PLA follower piston.

- When in automatic mode, both the PLA follower piston and manual load piston are retracted from the metering valve lever. They are held in the retracted position by fuel pressure when th auto/manual changeover solenoid valve is closed (energized).

Once in MANUAL mode, the pilot will have complete control of NP/NR by flight control manipulation and the throttle on the collective. The fuel flow slew rate is hydro mechanically limited to provide proper responsiveness for helicopter operation and also prevents surge. Fuel flow will be a function of the pilot controlled fuel metering valve orifice size. Maximum Continuous Power will be available for all ambient conditions. MANUAL mode fuel flow is pressure altitude compensated to maintain an approximate constant horsepower, with a change in altitude without throttle adjustment by the pilot. Fuel flow in the MANUAL mode, however, is not temperature compensated. Because of this, there may be temperatures at which maximum fuel flow in MANUAL will not be sufficient to achieve Takeoff Power.

NOTE

In the event engine (NP) overspeed system is activated during transition to or operation in manual mode, the control system is designed to keep the engine running. Engine may oscillate between 112.5% and 118.5% NP until corrective action is taken with throttle and collective. Refer to paragraph 1-23-M.

3. FADEC MODE switch — Depress one time.

Depressing the FADEC MODE switch one time will mute the FADEC FAIL warning horn (chime tone). As the transition to MANUAL MODE was initiated by the FADEC system, this step should not be accomplished until pilot is firmly established in MANUAL control.

This will allow pilot to keep hands on flight controls during the transition to MANUAL mode.

If the FADEC ECU is operational, it will track HMU operation, perform diagnostics, monitor engine functions and provide overspeed limiting for both NP and NG. Surge detection and avoidance will not be available. If an engine surge is encountered, decrease the throttle until the surge condition clears, then slowly increase the throttle to the desired power level. Rapid power changes should be avoided.

4. Land as soon as practical.

Applicable maintenance action will be required prior to next flight.

5. Normal shutdown if possible.

If normal shutdown can not be completed by rolling throttle to closed position, fuel shutoff valve can be positioned to off.

1-23-J. CATEGORY 2 - FADEC DEGRADED

FADEC DEGRADED faults represent a loss of some feature of the FADEC system which may cause a degradation in performance. This may result in NR droop, NR lag, or reduced maximum power capability. These faults will be displayed immediately when detected by the ECU. Operations should be continued in AUTO mode and helicopter is to be flown smoothly and nonaggressively. In conjunction with the FADEC DEGRADED light, the RESTART FAULT light may also activate under certain fault conditions.

Applicable maintenance action will be required prior to next flight.

1-23-K. CATEGORY 3 - FADEC FAULT

FADEC FAULT indicates that PMA and/or MGT, NP, or NG automatic limiting circuit(s) may not be functional. In conjunction with activation of the FADEC FAULT light, the RESTART FAULT light may also activate under certain fault conditions. These faults will be displayed immediately when detected by the ECU. Operations should continue in AUTO mode. If both lights (FADEC FAULT and RESTART FAULT) are illuminated, this indicates the MGT automatic limiting circuit (905°C in flight), may not be functional. The pilot should follow the appropriate procedures as set out in the Flight Manual.

Applicable maintenance action will be required prior to next flight.

1-23-L. CATEGORY 4 - RESTART FAULT

RESTART FAULT indicates a subsequent automatic engine start may not be possible. The fault does not require immediate action by the pilot and should not affect performance of the helicopter. It is recommended that the pilot plan the landing site accordingly. These faults will be displayed immediately when detected by the ECU and displayed as RESTART FAULT. Do not attempt a subsequent start until applicable maintenance action has been completed.

If the engine shutdown procedures are not properly followed, the manual mode pistons may begin to engage during the shutdown. The FADEC may then be unable to prevent a hot start on the next start and will indicate a RESTART FAULT to warn the pilot. An HMU manual piston parking procedure will be required, as described in paragraph 1-23-Q and the Rolls-Royce Allison 250-C47B Operations and Maintenance Manual.

1-23-M. ENGINE OVERSPEED PROTECTION

NP overspeed limiting is available in both the AUTO and MANUAL modes by independent analog circuits integral to the ECU. NG overspeed limiting is available in the AUTO and MANUAL modes through software control in the ECU. In the event of a FADEC FAILURE, it is possible that NG overspeed protection will not be available.

The FADEC ECU continuously monitors for NG and NP, or NP versus Torque (Q) overspeed conditions in both AUTO and MANUAL Mode.

Activation of the ENGINE OVSPD warning light will occur in the event of an NP overspeed, NG overspeed, or if NP verses Torque is above the continuous limit (102.4% NP at 100% Torque to 108.6% NP at 0% Torque). The light will also momentarily illuminate during the overspeed system test when the overspeed solenoid valve closes.

If the ENGINE OVSPD light is activated during engine operation due to an exceedance, and the value has been recorded by the ECU, the pilot will be provided with a maintenance advisory on shutdown in the form of a FADEC DEGRADED light. The FADEC DEGRADED light will illuminate when NG speed decays below 9.5%. If the pilot fails to recognize illumination of the FADEC DEGRADED light on shutdown, it will be illuminated the next time electrical power is applied following the FADEC system self test. When the FADEC DEGRADED light is illuminated as a maintenance advisory, maintenance investigation is required prior to further flight. Peak values of exceedances are located on the Engine History Data page of the EMC-35A Maintenance Terminal.

If limits are exceeded. Refer to 250-C47B Series Overspeed Limits in the Allison Operation and Maintenance Manual and Chapter 5 of BHT-407-MM-1.

- NP OVERSPEED

  When the engine reaches 118.5±1% NP, overspeed limiting will occur. The analog overspeed limiting feature will activate the overspeed solenoid valve which reduces fuel to the engine to a minimum flow condition (sub-idle value of 34 to 45 pph). The minimum fuel flow increases the likelihood of the engine remaining running and recovering from the overspeed. Once the NP speed drops to 112.5±2%, the overspeed solenoid valve will be deactivated and fuel flow will return to its previously commanded value.

In the event the overspeed can not be controlled after fuel flow is reintroduced, the overspeed limiting feature will control the overspeed between the activation trip point of 118.5±1% NP and the deactivation point of 112.5±2% NP. If this occurs, attempt to control engine and rotor speed with throttle and collective. Refer to the ENGINE OVERSPEED procedure in the Flight Manual.

- NG OVERSPEED

In AUTO mode, a software implemented overspeed system is provided. Should the software detect an NG overspeed, the protection feature will be activated. In addition, if the ECU has not failed, NG overspeed protection will be available in MANUAL mode.

When the engine reaches 110±1% NG, the ENGINE OVSPD warning light will illuminate and overspeed limiting will occur. The software controlled overspeed limiting feature will activate the overspeed solenoid valve which reduces fuel to the engine to a minimum flow condition (sub-idle value of 34 to 45 pph). The minimum fuel flow increases the likelihood of the engine remaining running and recovering from the overspeed. Once the NG speed drops to 107±1%, the overspeed solenoid valve will be deactivated and fuel flow will return to its previously commanded value.

In the event the overspeed cannot be controlled after fuel flow is reintroduced, the overspeed limiting feature will control the overspeed between the activation point of 110±1% NG and the deactivation point of 107±1% NG. If this occurs, attempt to control engine and rotor speed with throttle and collective. Refer to the ENGINE OVERSPEED procedure in the Flight Manual.

- OVERSPEED SYSTEM SHUTDOWN TEST

Functionality of the overspeed system is checked during FADEC power up and thereafter continuously by the ECU. Operation of the overspeed solenoid is checked periodically by the pilot through the use of the OVERSPEED SHUTDOWN test procedure.

The OVERSPEED SHUTDOWN test procedure will shut down the engine only if collective pitch is below 10%, throttle position is at idle, NG is between 60-66% and NP is less than 75%. The OVERSPEED test button must be pressed and held for a minimum of 1.0 second but not more than 10.0 seconds. Once the test button is released, the OVERSPEED test is completed as follows. The FADEC ECU signals the overspeed solenoid valve to close and the ENGINE OVSPD light to come on. Once the FADEC ECU senses an NG decrease greater than 0.5%, the overspeed solenoid valve is opened, the ENGINE OVSPD light goes off, and the engine is shut down by FADEC ECU activation of the hot start abort feature.

If the overspeed test is unsuccessful, the engine will continue to operate at idle power, the FADEC FAULT caution light will illuminate, and a normal shutdown procedure must be carried out.

1-23-N. DELETED

1-23-P. ENGINE SHUTDOWN

Pilot control of engine speed from 100% NP/NR to idle in AUTO mode is controlled through throttle movement. As the throttle is positioned from the detented FLY position to idle, electrical signals are sent to the ECU from the HMU - PLA potentiometer. These signals dictate the amount of authority the ECU has to control maximum fuel flow (NG limiting), and in turn, engine speed. Therefore, as throttle is decreased, the maximum fuel flow that can be delivered to the engine is reduced by positioning the fuel metering valve to control engine NG speed/power.

In the unlikely event that a system fault occurs which does not allow a reduction in engine speed by positioning the throttle to idle, complete the 2 minute cool-down at 100% flat pitch. After the 2 minute cool-down, the engine is to be shutdown by rolling the throttle to the CLOSED position.

Pilot control of engine speed from 100% NP/NR to NG idle in MANUAL mode is controlled hydro mechanically through throttle movement. Idle speed in MANUAL mode may not stabilize at 63±1% NG. If this occurs, maintain idle speed at 63±1% NG with throttle.

Following the appropriate cool down period at idle, the engine may be shut down in either the AUTO or MANUAL mode by positioning the throttle to cutoff. This will close the mechanical fuel shut off valve within the HMU.

Do not reposition the throttle out of cutoff unless NG has decayed to zero. If the throttle is positioned out of cutoff prior to the NG speed decreasing through 9.5%, the FADEC IN-FLIGHT restart logic will introduce fuel and activate the igniter. This can cause a relight and possible over temperature condition. Refer to paragraph 1-29.

If relight occurs, the pilot must immediately position the throttle to the closed position and activate the starter.

Additionally, the pilot must also allow NG to decay to 0% in AUTO mode prior to positioning the battery switch to off. If this procedure is not followed, the MANUAL mode pistons may move hydro mechanically after electrical power is removed. This may cause a RESTART FAULT the next time power i applied to the FADEC ECU and a HMU manua piston parking procedure will be required pe paragraph 1-23-Q or as described in th Allison 250-C47B Operation and Maintenanc Manual.

1-23-Q. HMU MANUAL PISTON PARKING PROCEDURE

Starting with the HMU manual mode pistons i the wrong position may result in a hot start o the engine. When the pilot is not certain of th position of the pistons, or has received a Maintenance Advisory that the pistons are ou of position, the following procedure wil assure the pistons are in the correct positio (fully retracted) for engine starting.

1. Position throttle to cutoff.
2. Pull IGNITER circuit breaker.
3. BATT - ON.
4. Power up check - Complete.
5. FADEC Mode switch - MANUAL.
6. Motor the engine (with throttle in cutoff) for 10 seconds.
7. Wait for NG to decay to 0%.
8. FADEC Mode switch - AUTO.
9. Motor the engine (with throttle in cutoff) for an additional 10 seconds.
10. Wait for NG to decay to 0%.
11. BATT - OFF.
12. Push in IGNITER circuit breaker.

Continue prestart checklist.

1-23-R. CATEGORY 5, - MAINTENANCE ADVISORY, FADEC SYSTEM FAULTS - ENGINE SHUTDOWN

Maintenance Advisory Faults are those detected by the ECU that are considered minor in nature and are not communicated to the cockpit with the engine operating. The FADEC DEGRADED light serves as the maintenance advisory light. This light will be illuminated, upon engine shutdown, if any fault or exceedance has been detected during the last engine run or if a current fault exists. This will indicate that maintenance action is required prior to the next flight.

Maintenance advisory faults will display during shutdown when the throttle is placed in the CLOSED position and the NG speed decays below 9.5%. If the pilot misses the maintenance advisory on shutdown, it will illuminate at the next application of electrical power.

1-23-S. CHECKING FADEC FAULT CODES

As stated previously, faults can be displayed immediately via a FADEC FAIL, FADEC MANUAL, FADEC DEGRADED, FADEC FAULT, RESTART FAULT or by a combination of these lights. Maintenance Advisory faults will be displayed on shutdown via the FADEC DEGRADED light. In addition, faults are also used to identify exceedances.

Regardless if the fault light(s) were displayed in flight or at shutdown, maintenance action is required prior to further flight. Refer to BHT-407-MM-1 and Rolls-Royce Allison 250-C47B Operation and Maintenance Manual.

The preferred method of determining FADEC faults or exceedances is with the Chandler Evans EMC-35A Maintenance Terminal. The EMC-35A Maintenance Terminal (DOS or Windows version) is capable of providing information on Current Faults, Last Engine Run Faults, Accumulated Faults (Fault History screen) and NP overspeed exceedance information (Engine History screen). Refer to the applicable Chandler Evans Maintenance Terminal User Guide for operating instructions.

If a maintenance terminal is not available, identify faults or exceedances using the Maintenance Mode feature of the FADEC system. This feature allows operators to determine faults through a sequence of flashing light displays on the cockpit caution panel.

The caution panel fault display may be operated per paragraph 1-23-S-1.

1-23-S-1. ENGINE RUN FAULT CODES - PROCEDURE FOR VIEWING

NOTE

Displayed faults may be LAST ENGINE RUN or CURRENT faults. Following this procedure, refer to paragraph 1-23-S-2 to determine if faults are current.

1. Engine must be shutdown and the FADEC MODE switch positioned to MANUAL. Place the collective full down (below 10%) and the throttle in the cutoff position.

NOTE

If the throttle or collective is moved during the above procedure or the FADEC MODE switch is positioned to AUTO, the FADEC ECU will exit the fault code reporting mode.

2. Depress and release FADEC ECU maintenance button on the left hand side of the lower pedestal to enter the fault code reporting mode.
3. FADEC DEGRADED, FADEC FAULT and the RESTART FAULT lights will simultaneously flash five times to indicate that the maintenance mode has been entered by the ECU.
4. Depress and release the FADEC ECU maintenance button. If a fault is present, it will be displayed by a specified number of FADEC DEGRADED caution panel light segment flashes.
5. Depress and release FADEC/ECU maintenance button to flash the next fault code.
6. Steady illumination of the FADEC DEGRADED caution panel light segment indicates that no other faults exist for this light.

7. Continue to depress and release the FADEC/ECU maintenance button to step through the FADEC FAULT and RESTART FAULT caution panel light segments as above. This will determine if fault codes exist for these segments.

8. If no fault code exists for the selected caution panel segment, the caution light will illuminate continuously when the FADEC/ECU maintenance button is released.

9. When interrogation is complete, the next push of the FADEC maintenance button will cause the FADEC DEGRADED, FADEC FAULT and RESTART FAULT caution light segments to flash simultaneously five times and then extinguish. This indicates that the FADEC ECU has exited the maintenance mode.

10. To determine fault code name from caution panel flashing display, refer to Table 1-2.

1-23-S-2 FADEC FAULT CODES — PROCEDURE TO DETERMINE LAST ENGINE RUN FAULTS FROM CURRENT FAULTS

The procedure to determine if the fault codes displayed are LAST ENGINE RUN faults or CURRENT faults may be accomplished by performing the steps listed in paragraph 1-23-S-1 with the throttle in the IDLE position.

With the throttle positioned to IDLE, any FADEC fault code which is displayed will be a current fault.

In addition, if no FADEC related lights are displayed on the caution, warning, advisory panel with electrical power applied, the FADEC in AUTO mode, and the throttle positioned to idle, no current faults exist.

1-23-S-3. FAULT CODE CHARTS - USE OF

If fault codes have been displayed via the caution panel, refer to Table 1-2 for specific information. This information can be used in conjunction with the Bell Helicopter Maintenance Manual BHT-407-MM-1 and the Rolls-Royce Allison Operation and Maintenance Manual 250-C47B. These publications contain the data to determine the maintenance action required prior to further flight.

For specific fault information displayed on the EMC-35A Maintenance Terminal, refer to the Rolls-Royce Allison 250-C47B Operation and Maintenance Manual to determine the maintenance action required prior to further flight.

NOTES

Item No.	Fault Code Name	Description	FADEC DEGRADED Indicator # Flashes	FADEC Fault Indicator # Flashes	RESTART FAULT Indicator # Flashes
1	FADEC	Control System Failure	1		
3	NPQ Exceedanc®	NP and Q (TORQUE)> Line 2	3		
5	MGT	MGT Indication Failure	5		
7	HMU — Auto/ Manual Solenoid	Failure to Control the Auto/Manual Solenoid	7		
8	CIT	CIT Temperature Indication Failure	8		
9	Open Metering Valve	Metering Valve not in Starting Position	9		
10	Starter Relay	Starter will not stay engaged	10		
11	Nr	Rotor Decay Anticipation	11		
12	Overspeed Test Switch	Incorrect Overspeed Switch Indication	12		
13	HMU	Failure to Control Fuel Flow		1	
15	NPQ Run Limit	NP and Q (TORQUE)> Line 1		3	
16	HMU Metering Valve	Failure in Metering Valve Position Reading		4	
17	CP	Collective Pitch Indication Failure		5	
18	HMU Stepper Motor	Failure to Control Stepper Motor		6	
20	HMU O/S Solenoid	Failure to Control O/S Solenoid		8	
21	Surge	Engine Surge Event		9	
22	NG	NG Speed Indication Failure		10	
23	Airframe Power	Airframe Power Supply Failure		11	
24	Overspeed	Overspeed Flag Set		12	
26	TMOP	Torque Indication Failure			2
29	HMU - PLA Position	Failure in PLA Position Reading			5
30	PMA	PMA Power Supply Failure			6
31	HMU Start Fuel Valve	Failure to Control the Hot Start Abort Solenoid			7
33	Ignition Relay	Failure to Control the Ign Relay			9
34	NP	NP Speed Indication Failure			10
35	AMSwFlt	Incorrect Auto/Manual Switch Indication			11
36	QMSwFlt	Quiet Mode Switch Fault			12
37		All Faults Have Been Displayed	ON	ON	ON

igure 5-5 Software version 5202 Fault code display

1-23-T. CLEARING FADEC FAULT CODES

Faults / exceedances are not to be erased unless appropriate maintenance actions have been carried out in accordance with 407 maintenance manual and Rolls-Royce Allison 250-C47B operations and maintenance manual. Do not attempt to clear any fault or exceedance while the engine is operating.

If maintenance actions have been conducted due to a recorded exceedance or to correct a current or last engine run fault, it must be ensured that no FADEC system lights are illuminated when the throttle is positioned to idle for the next start attempt. If a FADEC related light is illuminated with the throttle positioned to idle, a current fault exists and further maintenance action is required.

1-23-T-1. CURRENT FAULTS

Current faults may be cleared by performing a power reset (Battery switch OFF/ON). If fault is no longer detected, associated FADEC DEGRADED light will be extinguished.

1-23-T-2. LAST ENGINE RUN FAULTS / EXCEEDANCES

LAST ENGINE RUN faults or Exceedances may be cleared by performing a successful engine start or with use of the EMC-35A Maintenance Terminal.

To erase LAST ENGINE RUN Faults / Exceedances, refer to the EMC-35A maintenance terminal Users Guide for operating instructions.

1-23-T-3. ACCUMULATED FAULTS / EXCEEDANCES

Accumulated faults or Exceedances may only be cleared with use of the EMC-35A Maintenance Terminal.

NP Exceedance values may only be cleared from engine history data page with use of the EMC-35A Maintenance Terminal.

1-23-U. FADEC TRAINING IN MANUAL MODE

Prior to actual flight training in the MANUAL mode, an operational understanding of the FADEC system along with a sound knowledge of emergency procedures is required. It is recommended that training be first accomplished in the helicopter in a cockpit procedural environment (engine not running). This can provide a visual simulation of the horn and lights associated with a FADEC failure direct to MANUAL mode and familiarize the pilot with the required cockpit actions. This should be followed by a takeoff/hover/ circuit and landing in MANUAL mode which will allow the pilot to become familiar with the required manipulation of the throttle and controls.

Once the pilot is comfortable with flight in MANUAL mode, simulated FADEC failure emergency procedures can be carried out in flight.

1-23-V. COCKPIT PROCEDURAL TRAINING ON GROUND (ENGINE NOT RUNNING)

1. Pull START and IGNTR circuit breakers and ensure fuel valve switch is positioned to OFF.
2. Connect external power source.
3. Allow instruments and FADEC to complete self test with FADEC MODE switch in AUTO.
4. Position throttle to FLY detent and collective to approximate cruise flight setting.
5. Simulate FADEC FAILURE by pulling FADEC circuit breaker. This will simulate a failure direct to MANUAL mode. FADEC FAIL warning horn will activate along with illumination of the FADEC FAIL and FADEC MANUAL caution panel lights. FADEC MODE switch will illuminate MAN.

6. Carry out the appropriate Flight Manual emergency response procedure. (Depressing the FADEC MODE switch will silence the horn).
7. Push in FADEC circuit breaker and set FADEC MODE switch to AUTO.
8. Repeat procedure until it is understood.
9. Disconnect external power source, position throttle to cutoff and push in START and IGNTR circuit breakers.

1-23-W. FLIGHT TRAINING IN MANUAL MODE

In MANUAL MODE, the following is applicable:

— Switching to MANUAL mode at idle (63±1%NG) may result in change in NG.

— Igniter operates continuously.

— AUTO RELIGHT, FADEC MANUAL caution panel lights illuminate.

— FADEC MODE switch indicates MAN.

— FADEC ECU remains operational, however, surge protection and avoidance logic is not available. In the event of a FADEC FAILURE while training in MANUAL mode, the FADEC FAIL warning light will illuminate. FADEC FAIL horn will not sound. Remain in MANUAL and land as soon as practical.

— Maximum Continuous Power is available for all ambient conditions. Takeoff Power may not be available.

1. Perform START procedure and SYSTEMS CHECKS in AUTO mode.
2. At idle (63±1% NG), depress FADEC MODE switch to transition to MANUAL mode.

NOTE

Transition back to AUTO mode can be made at any time by depressing FADEC MODE switch to AUTO. Upon selecting AUTO mode, an engine power transient may be experienced as the FADEC ECU matches engine power to rotor load. Ensure throttle is positioned to FLY detent position following selection of AUTO mode.

3. Manipulate throttle on ground to become familiar with MANUAL control.
4. Increase throttle to maintain 95 to 100% NP/NR. Manipulate throttle/ collective and lift into hover.
5. When comfortable with manipulation of throttle and flight controls in hover, transition into forward flight and conduct circuits to touchdown.

1-23-X. SIMULATED FADEC FAILURE TRAINING (IN FLIGHT)

When comfortable with flying circuits to touch down in MANUAL mode, transitions between AUTO and MANUAL may be conducted in flight.

Simulation can be accomplished by activating the FADEC FAIL horn by pushing the FADEC FAIL horn test button. Although the applicable FADEC caution panel lights cannot be activated, the pilot should respond to the FADEC FAIL horn and carry out the Flight Manual emergency response procedure. Once the throttle bezel position is matched to the actual NG indication, pilot shall position the fadec mode switch to MANUAL.

Following simulated FADEC FAILURE training in flight, ensure FADEC MODE switch is positioned to AUTO prior to shutdown. This will ensure the manual pistons in the HMU are parked, and a subsequent start in AUTO can be carried out without maintenance action.

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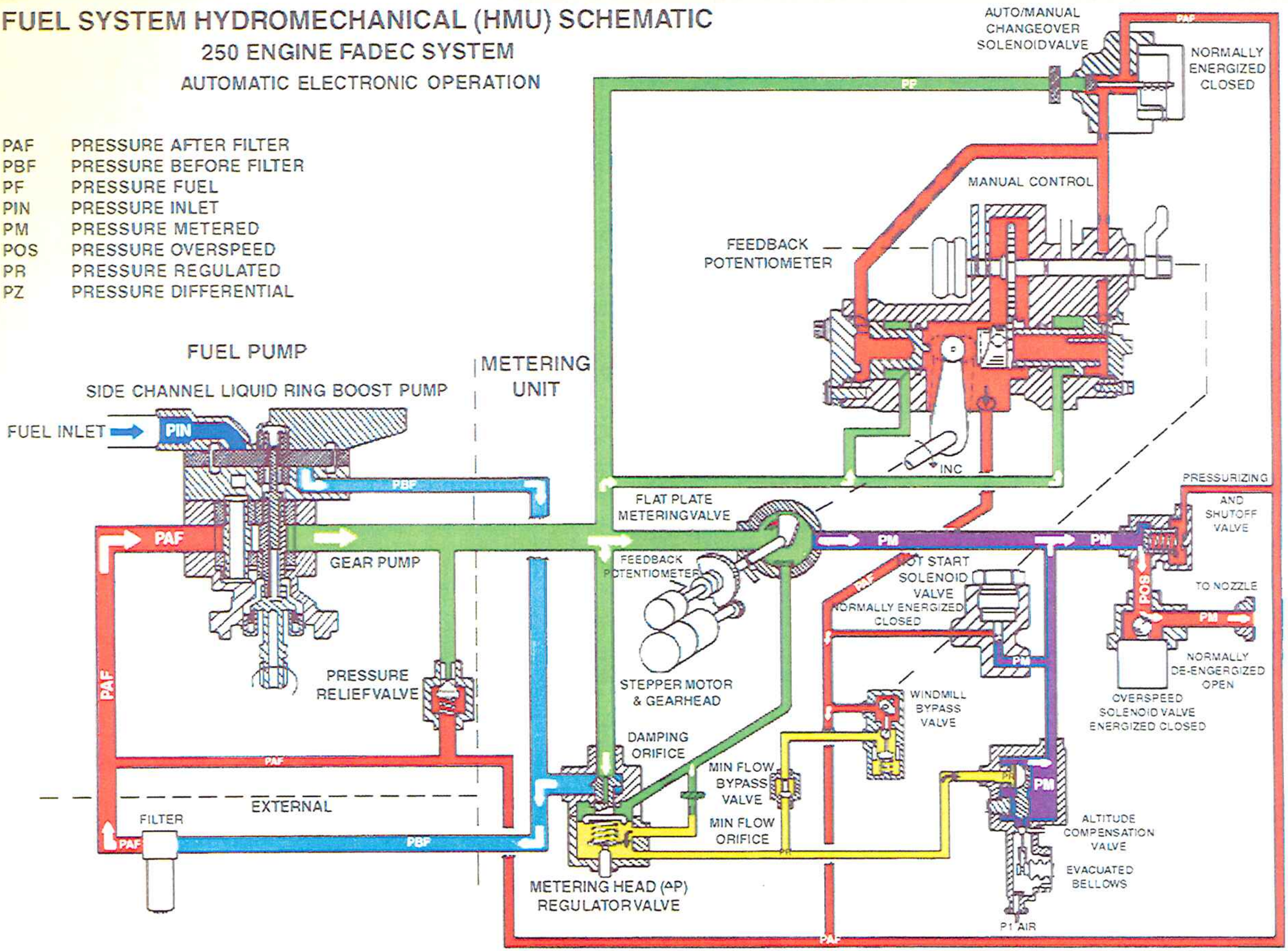
FUEL SYSTEM HYDROMECHANICAL (HMU) SCHEMATIC

250 ENGINE FADEC SYSTEM

AUTOMATIC ELECTRONIC OPERATION

PAF	PRESSURE AFTER FILTER
PBF	PRESSURE BEFORE FILTER
PF	PRESSURE FUEL
PIN	PRESSURE INLET
PM	PRESSURE METERED
POS	PRESSURE OVERSPEED
PR	PRESSURE REGULATED
PZ	PRESSURE DIFFERENTIAL

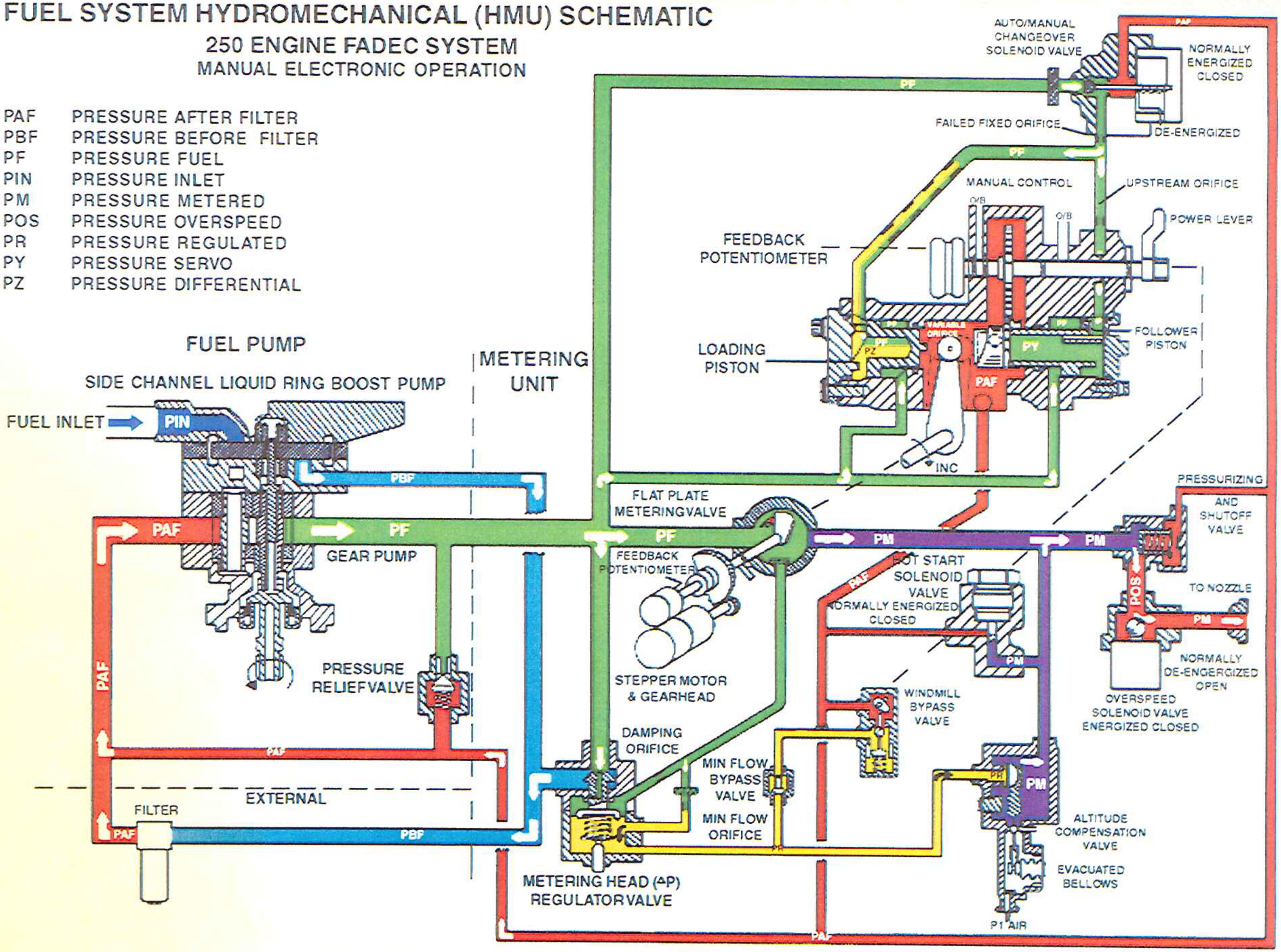


FUEL SYSTEM HYDROMECHANICAL (HMU) SCHEMATIC

250 ENGINE FADEC SYSTEM

MANUAL ELECTRONIC OPERATION

PAF	PRESSURE AFTER FILTER
PBF	PRESSURE BEFORE FILTER
PF	PRESSURE FUEL
PIN	PRESSURE INLET
PM	PRESSURE METERED
POS	PRESSURE OVERSPEED
PR	PRESSURE REGULATED
PY	PRESSURE SERVO
PZ	PRESSURE DIFFERENTIAL



ELECTRONIC CONTROL UNIT

The **ELECTRONIC CONTROL UNIT (ECU)** is mounted on four vibration isolators, on the roof of the aircraft in front of the transmission. The **ECU** controls, and monitors numerous internal and external inputs to modulate fuel flow to the engine while maintaining helicopter rotor speed. There is an ambient pressure sensor port on the front side next to the cannon plugs, used to sense pressure altitude changes. The **ECU** offers numerous advantages over non-electronic fuel controlled aircraft during starting, in-flight, shutdown, safety and maintainability.

HYDROMECHANICAL UNIT (HMU)

The **HYDROMECHANICAL UNIT (HMU)** is a combination of fuel pump and fuel control, designed to pump and meter fuel to a turbine engine.

A dual element fuel pump, fuel metering valve, manual fuel control, power lever input shaft, and two feedback potentiometers are some of the components within the HMU.

The dual element fuel pump is mounted on the right rear accessory gearbox drive pad. The pump is turning and pumping any time the Ng gear train is turning. The pump output volume is approximately 4.5 gallons per minute (GPM). A pressure relief valve limits pump output pressure to 900 PSI above inlet pressure.

MONOPOLE POLE PICKUPS

Monopole pickups utilize an induction principle. Each monopole pickup is mounted very close to and points to a gear, or castleated washer which begins to rotate with the engine. **EXAMPLE**: As a gear rotates in front of the monopole pickup, one tooth generates an electronic field within the monopole pickup, as the gear rotates, a space appears in front of the monopole pickup which collapses the field. Each build-up and collapse of electronic field sends an electronic signal to an appropriate gage or sensor read by the pilot (for example Ng.).

There are three monopole pickups used in the 407. Two are mounted on the top right side of the accessory gearbox, and sense the engine Ng and Np speeds. The third is mounted on the right rear side of the transmission, and senses the Nr.

MEASURED GAS TEMPERATURE (MGT)

Four probes mounted and equally spaced around the engine sense the temperature of passing gasses between Ng and Np turbine wheels. Each thermocouple generates a DC voltage which is directly proportional to the gas temperature it senses. The average of the four probes is a signal to the MGT gage, and ECU.

COMPRESSOR INLET AIR TEMPERATURE SENSOR (CIT)

The sensor is mounted on the left side of the forward engine firewall bulkhead. The **CIT** probe is mounted such that air entering the engine intake and plenum chamber will pass over the probe. The probe provides an ambient air temperature signal to the ECU, for **DENSITY ALTITUDE (Hd)** calculations.

PERMANENT MAGNET ALTERNATOR (PMA)

The **PMA** is an engine driven self generating electrical device which provides electrical power **ONLY TO THE ECU**, and its components, at Np speeds of 85% and above.

COMBINED ENGINE FILTER ASSEMBLY (CEFA)

Two filters co-located on the rear left of the accessory gearbox, house a second in line fuel filter, and a scavenge engine oil filter. Both filters have red pop out type impending bypass indicators (PSID), for service indications.

COLLECTIVE PITCH TRANSDUCER

The collective pitch transducer is an electrical potentiometer installed under the copilot seat. Mounted on one end to the airframe, and the other end to the collective jackshaft. With each movement of the collective (up or down), the ECU receives preliminary information and will make fuel flow adjustments necessary to prevent any overspeed or droop of the power turbine. The collective potentiometer also informs the ECU of collective rate of movement and collective position.

ADDITIONAL COMPONENTS

The **FADEC** system includes an igniter relay, starter relay, torque meter oil transducer, 28v DC airframe power source, FADEC ECU maintenance port, and cockpit displays.

NOTES