A1-H60BB-NFM-000

NATOPS FLIGHT MANUAL
NAVY MODEL
SH-60B HELICOPTER

THIS PUBLICATION SUPERSEDES
A1-H60BB-NFM-000 DATED 15 JANUARY 2006.

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NAVAL AIR SYSTEMS COMMAND
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47123 BUSE ROAD, BLDG 2272
PATUXENT RIVER, MD 20670-1547

01 APRIL 2008

LETTER OF PROMULGATION

1. The Naval Air Training and Operating Procedures Standardization (NATOPS) Program is a positive approach toward improving combat readiness and achieving a substantial reduction in the aircraft mishap rate. Standardization, based on professional knowledge and experience, provides the basis for development of an efficient and sound operational procedure. The standardization program is not planned to stifle individual initiative, but rather to aid the commanding officer in increasing the unit's combat potential without reducing command prestige or responsibility.

2. This manual standardizes ground and flight procedures but does not include tactical doctrine. Compliance with the stipulated manual requirements and procedures is mandatory except as authorized herein. In order to remain effective, NATOPS must be dynamic and stimulate rather than suppress individual thinking. Since aviation is a continuing, progressive profession, it is both desirable and necessary that new ideas and new techniques be expeditiously evaluated and incorporated if proven to be sound. To this end, commanding officers of aviation units are authorized to modify procedures contained herein, in accordance with the waiver provisions established by OPNAVINST 3710.7, for the purpose of assessing new ideas prior to initiating recommendations for permanent changes. This manual is prepared and kept current by the users in order to achieve maximum readiness and safety in the most efficient and economical manner. Should conflict exist between the training and operating procedures found in this manual and those found in other publications, this manual will govern.

3. Checklists and other pertinent extracts from this publication necessary to normal operations and training should be made and carried for use in naval aircraft.

S. R. EASTBURG
Rear Admiral, United States Navy
By direction of
Commander, Naval Air Systems Command

INTERIM CHANGE SUMMARY

The following Interim Changes have been cancelled or previously incorporated into this manual.

INTERIM CHANGE NUMBER(S)	REMARKS/PURPOSE
1 thru 63	Previously Incorporated

The following Interim Changes have been incorporated into this Change/Revision.

INTERIM CHANGE NUMBER(S)	REMARKS/PURPOSE
64	Misc. H-60 Common Changes
65	Misc. Changes
66	Fuel Prime Checklist Procedure
67	Changes to Practice Autorotative Approach

Interim Changes Outstanding — To be maintained by the custodian of this manual.

INTERIM CHANGE NUMBER	ORIGINATOR/DATE (or DATE/TIME GROUP)	PAGES AFFECTED	REMARKS/PURPOSE
68	202000Z MAR 09	3-17, 7-1, 7-21, 7-32, 7-57, 8-15, 10-13, 10-33	Tiedown Requirements
69	262003Z AUG 09	10-67	Revised Auto RPM Correct Chart (and re-issue of IC-68 for the Conference Revision Release).
70	052000Z OCT 09	2-33/34, 7-21/32, 8-38, 9-13/ 14/ 15/27/28, 10-16/19/67	New Operational Warnings & Engine Diaphragm Coupling Caution, correction of Autorotation RPM Correction Chart.

P 052000Z OCT 09
FROM COMNAVAIRSYSCOM PATUXENT RIVER MD//4.0P//
TO ALL SEAHAWK HELICOPTER ACTIVITIES
INFO COMNAVAIRSYSCOM PATUXENT RIVER MD//4.0P/4.1/4.1.1.2/5.0F/5.1//
PEOASWASM PATUXENT RIVER MD//PMA299//
COMNAVAIRFOR SAN DIEGO CA//N455/N3C3/N421L//
COMNAVAIRFORES SAN DIEGO CA//N42/N52/N421D1//
COMNAVSAFECEN NORFOLK VA//11//
COMNAVSURFLANT NORFOLK VA//N42/N420//
COMNAVSURFPAC SAN DIEGO CA//N42//
FLTREADCENSOUTHEAST JACKSONVILLE FL//3.3.3//
HMX ONE QUANTICO VA//C148-11//
MSGID/GENADMIN/COMNAVAIRSYSCOM/4.0P//
SUBJ/SH-60B AIRCRAFT NATOPS PUBLICATIONS INTERIM CHANGE//
REF/A/DESC:DOC/COMNAVAIRFOR/25SEP2009//
REF/B/DESC:DOC/COMNAVAIRSYSCOM/03SEP2009//
REF/C/DESC:NA A1-H60BB-NFM-000/COMNAVAIRSYSCOM/01APR2008//
REF/D/DESC:NA A1-H60BB-NFM-500/COMNAVAIRSYSCOM/01APR2008//
REF/E/DESC:NA A1-H60BB-NFM-700/COMNAVAIRSYSCOM/01APR2008//
NARR/REF A IS SH-60B COG COMMAND CONCURRENCE.
REF B IS AIRS NATOPS-2009-134 & 173.
REF C IS SH-60B NATOPS FLIGHT MANUAL (NFM).
REF D IS SH-60B NATOPS PILOTS POCKET CHECKLIST (PPCL).
REF E IS SH-60B NATOPS FUNCTIONAL CHECKFLIGHT CHECKLIST (FCFCL).//
GENTEXT/REMARKS/1. THIS MESSAGE IS ISSUED IN RESPONSE TO REFS A AND B. THIS MESSAGE ISSUES INTERIM CHANGE (IC) NUMBER 70 TO REF C, IC NUMBER 64 TO REF D, AND IC NUMBER 23 TO REF E.

2. SUMMARY.
   A. THIS MESSAGE INCORPORATES NEW OPERATIONAL WARNINGS AND ENGINE DIAPHRAGM COUPLING CAUTIONS IN REFS C AND E. THIS MESSAGE ALSO MAKES TYPOGRAPHIC (REFS C AND D) AND RPM CHART (REF C) CORRECTIONS INADVERTENTLY RELEASED WITH THE LAST IC.
   B. REPLACEMENT PAGES CONTAINING THESE CHANGES FOR DOWNLOADING AND INSERTION INTO REFS C, D, AND E WILL BE ATTACHED TO THIS INTERIM CHANGE MESSAGE WHEN IT IS POSTED ON THE NATEC AND AIRWORTHINESS WEBSITES (SEE LAST PARA BELOW).
3. THESE CHANGES IMPACT THE FOLLOWING NATOPS CHECK LISTS. THE REPLACEMENT PAGE PACKAGE INCLUDES THE FOLLOWING PAGES:
   A. REF C (SH-60B NFM -000) 5/(6 BLANK), 2-33/34, 7-21/22, 7-31/32, 8-37/38, 9-13 TO 16, 9-27/28, 10-15/16, 10-19/20, AND 10-67/68.
   B. REF D (SH-60B PPCL-500) PAGES B/(C BLANK), 29 AND 30.
   C. REF E (SH-60B FCFCL-700) PAGES B/(C BLANK), 1-23 TO 28.
4. POINTS OF CONTACT:
   A. SH-60B NATOPS PROGRAM MANAGER:LT JAKE HAFF, HSL-40, TEL DSN 270-6332, OR COMM (904)270-6332 x222, EMAIL:JAKE.L.HAFF1@NAVY.MIL.
   B. NAVAIR POCS:
      (1) MARTY SCANLON, NATOPS IC COORDINATOR, TEL DSN 757-6045 OR COMM (301) 757-6045, EMAIL: MARTIN.SCANLON@NAVY.MIL
      (2) LCDR WADE HARRIS, AIR-4.1.1.2, IN-SERVICE H-60 CLASS DESK, DSN 757-5343 OR (301)757-5343, EMAIL:BENJAMIN.W.HARRIS@NAVY.MIL.
      (3) LCDR BEN KELSEY, 4.0P NATOPS OFFICER, DSN 995-2502, COM 301-995-2505, EMAIL: BEN.KELSEY@NAVY.MIL.

(4) AIRWORTHINESS GLOBAL CUSTOMER SUPPORT TEAM,
(301) 757-0187, EMAIL AIRWORTHINESS@NAVY.MIL.

5. OTHER REMARKS:

A. THIS MESSAGE WILL BE POSTED ON THE AIRWORTHINESS WEBSITE, HTTPS:AIRWORTHINESS.NAVAIR.NAVY.MIL WITHIN 48 HOURS OF RELEASE. INTERIM CHANGES MAY BE FOUND IN TWO PLACES ON THE WEBSITE:
(1) IN THE NATOPS LIBRARY SORTED BY AIRCRAFT PLATFORM AND TMS.
(2) IN AIRS, SEARCH BY AIRS NUMBER FOUND IN REF B ABOVE.

B. THIS MESSAGE WILL ADDITIONALLY BE POSTED ON THE NATEC WEBSITE, WWW.MYNATEC.NAVY.MIL. IF THE IC MESSAGE INCLUDES REPLACEMENT PAGES, THEY WILL BE PLACED WITHIN THE MANUAL AND REPLACED PAGES DELETED. IF UNABLE TO VIEW THIS MESSAGE ON EITHER THE AIRWORTHINESS OR NATEC WEBSITES, INFORM THE NATOPS GLOBAL CUSTOMER SUPPORT TEAM AT (301) 342-3276, DSN 342-3276, OR BY EMAIL AT NATOPS@NAVY.MIL.

C. INFORMATION REGARDING THE AIRWORTHINESS PROCESS, INCLUDING A LISTING OF ALL CURRENT INTERIM FLIGHT CLEARANCES, NATOPS AND NATIP PRODUCTS ISSUED BY NAVAIR 4.0P, CAN BE FOUND AT OUR WEBSITE: HTTPS:(SLASH)(SLASH)AIRWORTHINESS.NAVAIR.NAVY.MIL.

D. EPOWER FOLDER ID 876128, TRACKING NUMBER 35443.//

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BEN KELSEY, NATOPS OFFICER, 4.0P, 10/05/2009

P 262003Z AUG 09
FROM COMNAVAIRSYSCOM PATUXENT RIVER MD//4.0P//
TO ALL SEAHAWK HELICOPTER ACTIVITIES
INFO COMNAVAIRSYSCOM PATUXENT RIVER MD//4.0P/4.1/4.1.1.2/5.0F/5.1//
PEOASWASM PATUXENT RIVER MD//PMA299//
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FLTREADCENSOUTHEAST JACKSONVILLE FL//3.3.3//
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MSGID/GENADMIN/COMNAVAIRSYSCOM/4.0P//
SUBJ/SH-60B AIRCRAFT NATOPS PUBLICATIONS INTERIM CHANGE//
REF/A/DESC:DOC/COMNAVAIRFOR/08JUN2009//
REF/B/DESC:DOC/COMNAVAIRSYSCOM/28APR2009//
REF/C/DESC:DOC/COMNAVAIRSYSCOM/17MAR2009//
REF/D/DESC:DOC/COMNAVAIRSYSCOM/05SEP2008//
REF/E/DESC:NA A1-H60BB-NFM-000/COMNAVAIRSYSCOM/01APR2008//
REF/F/DESC:NA A1-H60BB-NFM-500/COMNAVAIRSYSCOM/01APR2008//
REF/G/DESC:NA A1-H60BB-NFM-700/COMNAVAIRSYSCOM/01APR2008//
REF/H/DESC:NA A1-H60BB-NFM-800/COMNAVAIRSYSCOM/01APR2008//
NARR/REF A IS SH-60B COG COMMAND CONCURRENCE.
REF B IS AIRS NATOPS-2009-059.
REF C IS AIRS NATOPS-2009-051.
REF D IS AIRS NATOPS-2008-136.
REF E IS SH-60B NATOPS FLIGHT MANUAL (NFM).
REF F IS SH-60B NATOPS PILOTS POCKET CHECKLIST (PPCL).
REF G IS SH-60B NATOPS FUNCTIONAL CHECKFLIGHT CHECKLIST (FCFCL).
REF H IS SH-60B NATOPS AIRCREW POCKET CHECKLIST (APCL).//
GENTEXT/REMARKS/1. THIS MESSAGE IS ISSUED IN RESPONSE TO REFS A, B, C, AND D. THIS MESSAGE ISSUES INTERIM CHANGE (IC) NUMBER 69 TO REF E, IC NUMBER 63 TO REF F, IC NUMBER 22 TO REF G, AND IC NUMBER 11 TO REF H.

2. SUMMARY.
   A. THIS MESSAGE ADDRESSES CHANGES TO THE AUTOROTATION RPM CORRECTION CHART IN REFS E AND G, AND TO THE SONOBUOY LOADOUT QUICK MIX TABLE IN REF H.
   B. THIS MESSAGE ALSO RE-ISSUES THE TIEDOWN REQUIREMENTS IC CONTENT RELEASED VIA DTG 202002Z MAR 2009 TO THE POST CONFERENCE REVISION PUBLICATIONS.
   C. REPLACEMENT PAGES CONTAINING THESE CHANGES FOR DOWNLOADING AND INSERTION INTO REFS E THROUGH H WILL BE ATTACHED TO THIS INTERIM CHANGE MESSAGE WHEN IT IS POSTED ON THE NATEC AND NATOPS WEBSITES (SEE LAST PARA BELOW).

3. THESE CHANGES IMPACT THE FOLLOWING NATOPS CHECK LISTS. THE REPLACEMENT PAGE PACKAGE INCLUDES THE FOLLOWING PAGES:
   A. REF E (SH-60B NFM -000) PAGES 5/(6 BLANK), 3-17/18, 7-1/2, 7-21/22, 7-31/32, 7-57/58, 8-15/16, 10-13/14, 10-33/34,10-67, AND 10-68.
   B. REF F (SH-60B PPCL -500) PAGES B/(C BLANK), 11, 12, 29, 30, 33, 34, 45, 46, 223, AND 224.
   C. REF G (SH-60B FCFCL -700) PAGES TITLE, A, B/(C BLANK), 1-19/20, 1-45/46, 1-89/(1-90 BLANK), 2-4/5, AND 2-12.
   D. REF H (SH-60B APCL -800) PAGES B/(C BLANK) AND 69 THROUGH 70B.
   E. TO ENSURE THE PDF PAGES PRINT TO SCALE: SELECT PRINT

AND VIEWING PRINT SETUP WINDOW, ENSURE "NONE" IS
SELECTED IN THE PAGE SCALING DROPDOWN.

4. POINTS OF CONTACT:
   A. SH-60B NATOPS PROGRAM MANAGER: LT JAKE HAFF, HSL-40, TEL DSN 960-6332, OR COMM (904) 270-6332 X222,MAIL:JAKE.L.HAFF1@NAVY.MIL.
   B. NAVAIR POCS:
      (1) MARTY SCANLON, NATOPS IC COORDINATOR, TEL DSN 757-6045 OR COMM (301) 757-6045, EMAIL: MARTIN.SCANLON@NAVY.MIL
      (2) LCDR WADE HARRIS, AIR-4.1.1.2, IN-SERVICE H-60 CLASS DESK, DSN757-5343OR(301)757-5343, EMAIL:BENJAMIN.W.HARRIS@NAVY.MIL.
      (3) KRISTIN SWIFT, 4.0P NATOPS CHIEF ENGINEER, TEL DSN 995-4193 OR COMM (301) 995-4193, EMAIL: KRISTIN.SWIFT@NAVY.MIL.
      (4) LCDR BEN KELSEY, 4.0P NATOPS OFFICER, DSN 995-2502, COM 301-995-2505, EMAIL: BEN.KELSEY@NAVY.MIL.
      (5) AIRWORTHINESS GLOBAL CUSTOMER SUPPORT TEAM, (301)757-0187, EMAIL AIRWORTHINESS@NAVY.MIL.
5. OTHER REMARKS:
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      (1) IN THE NATOPS LIBRARY SORTED BY AIRCRAFT PLATFORM AND TMS.
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   C. INFORMATION REGARDING THE AIRWORTHINESS PROCESS, INCLUDING A LISTING OF ALL CURRENT INTERIM FLIGHT CLEARANCES, NATOPS AND NATIP PRODUCTS ISSUED BY NAVAIR 4.0P, CAN BE FOUND AT OUR WEBSITE: HTTPS:(SLASH)(SLASH)AIRWORTHINESS.NAVAIR.NAVY.MIL.
   D. EPOWER FOLDER ID 847995, TRACKING NUMBER 32776.//

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Ben Kelsey, NATOPS Officer, 4.0P, 08/27/2009

Summary of Applicable Technical Directives

Information relating to the following recent technical directives has been incorporated into this manual.

CHANGE NUMBER	DESCRIPTION	DATE INC. IN MANUAL	VISUAL IDENTIFICATION
AYC 1345	MISSION FLEXIBILITY KIT, Incorporation of; (RAMEC CHPT 33-03).	20 Dec 2005	

Information relating to the following applicable technical directives will be incorporated in a future change.

CHANGE NUMBER	DESCRIPTION	DATE INC. IN MANUAL	VISUAL IDENTIFICATION

RECORD OF CHANGES

Record entry and page count verification for each printed change and erratum:

Change No. and Date of Change	Date of Entry	Page Count Verified by (Signature)

LIST OF EFFECTIVE PAGES

Effective Pages	Page Numbers
Original	1 (Reverse Blank)
Original	3 (Reverse Blank)
Original	5 (Reverse Blank)
Original	7 (Reverse Blank)
Original	9 (Reverse Blank)
Original	11 (Reverse Blank)
Original	13 thru 41 (Reverse Blank)
Original	43 thru 53 (Reverse Blank)
Original	55 (Reverse Blank)
Original	1-1 thru 1-14
Original	2-1 thru 2-144
Original	3-1 thru 3-20
Original	4-1 thru 4-17 (Reverse Blank)
Original	57 (Reverse Blank)
Original	5-1 thru 5-7 (Reverse Blank)
Original	59 (Reverse Blank)
Original	6-1 thru 6-10
Original	7-1 thru 7-95 (Reverse Blank)
Original	8-1 thru 8-38
Original	9-1 thru 9-63 (Reverse Blank)
Original	10-1 thru 10-68
Original	61 (Reverse Blank)
Original	11-1 thru 11-20
Original	63 (Reverse Blank)
Original	12-1 thru 12-101 (Reverse Blank)

Effective Pages	Page Numbers
Original	65 (Reverse Blank)
Original	13-1 thru 13-3 (Reverse Blank)
Original	14-1 thru 14-9 (Reverse Blank)
Original	67 (Reverse Blank)
Original	15-1 thru 15-44
Original	16-1 thru 16-24
Original	69 (Reverse Blank)
Original	17-1 (Reverse Blank)
Original	18-1 (Reverse Blank)
Original	71 (Reverse Blank)
Original	19-1 thru 19-8
Original	73 (Reverse Blank)
Original	20-1 thru 20-29 (Reverse Blank)
Original	75 (Reverse Blank)
Original	21-1 (Reverse Blank)
Original	22-1 thru 22-11 (Reverse Blank)
Original	23-1 thru 23-6
Original	24-1 thru 24-2
Original	25-1 thru 25-5 (Reverse Blank)
Original	26-1 thru 26-2
Original	27-1 thru 27-8
Original	28-1 thru 28-25 (Reverse Blank)
Original	Index-1 thru Index-37 (Reverse Blank)

NATOPS Flight Manual

SH-60B

CONTENTS

Page No.

Page No.

Page No.

Page No.

CHAPTER 7 — NORMAL PROCEDURES

Page No.

Page No.

Page No.

CHAPTER 10 — FUNCTIONAL CHECKFLIGHT PROCEDURES

Page No.

PART IV — FLIGHT CHARACTERISTICS

CHAPTER 11 — AERODYNAMIC CHARACTERISTICS AND TECHNIQUES IN VARIOUS REGIMES

Page No.

Page No.

Page No.

LIST OF ILLUSTRATIONS

Page No.

CHAPTER 1 — GENERAL DESCRIPTION

CHAPTER 2 — SYSTEMS

Page No.

Page No.

CHAPTER 3 — SERVICING AND HANDLING

CHAPTER 4 — HELICOPTER OPERATING LIMITATIONS

CHAPTER 5 — AIRCREW TRAINING, QUALIFICATIONS, AND REQUIREMENTS

CHAPTER 6 — FLIGHT PREPARATION

CHAPTER 7 — NORMAL PROCEDURES

CHAPTER 8 — SHIPBOARD PROCEDURES

Page No.

CHAPTER 9 — SPECIAL PROCEDURES

CHAPTER 10 — FUNCTIONAL CHECKFLIGHT PROCEDURES

Page No.

CHAPTER 11 — AERODYNAMIC CHARACTERISTICS AND TECHNIQUES IN VARIOUS REGIMES

CHAPTER 12 — EMERGENCY PROCEDURES

CHAPTER 15 — COMMUNICATION EQUIPMENT AND PROCEDURES

Page No.

CHAPTER 16 — NAVIGATION

CHAPTER 20 — NATOPS EVALUATION

CHAPTER 22 — STANDARD DATA

CHAPTER 23 — TAKEOFF

CHAPTER 24 — CLIMB

Page No.

CHAPTER 25 — RANGE

CHAPTER 26 — HOVER ENDURANCE

CHAPTER 27 — EMERGENCY OPERATION

CHAPTER 28 — LEVEL FLIGHT PERFORMANCE CHARTS

LIST OF ABBREVIATIONS/ACRONYMS

A

A/A. Air-to-air.

A/C. Aircraft.

AC. Alternating current.

ACIP. Armament control-indicator panel.

ACP. Audio converter processor.

ACRT. Armament controller receiver-transmitter.

ACS. Air capable ship.

ADB. Aircraft discrepancy book.

ADF. Automatic direction finder.

ADHEELS. Advanced helicopter emergency egress lighting system.

ADIZ. Air defense identification zone.

ADS. Analyzer detection set.

AF. Adaptability/flexibility.

AFCS. Automatic flight control system.

AGB. Accessory gear box.

AGCA. Attitude gyroscope control assembly.

AGL. Above ground level.

AH SDC. Armed helo signal data converter.

AHS. Armed helo system.

AI. Attitude indicator.

ANI. Assistant NATOPS Instructor.

AOA. Angle of attack.

AOP. Avionics operational program.

APU. Auxiliary power unit.

ARA. Armament relay assembly.

AS. Assertiveness.

ASDC. Armament signal data converter.

ASE. Aircraft survivability equipment.

ASST. Anti-ship surveillance and targeting.

ASuW. Anti-surface warfare.

ASW. Anti-submarine warfare.

ATABS. Automated track and balance set.

ATF. Aircraft torque factor.

ATO. Airborne tactical officer.

AVT. Automatic video tracker.

B

BARALT. Barometric altimeter.

BDHI. Bearing-distance-heading indicator.

BHA. Battle hit assessment.

BIDS. Bridge indication display system.

BIM®. Blade Inspection Method.

BIT. Built-in-test.

BITE. Built-in test equipment.

BRC. Base recovery course.

BRKE. Brake.

BRU. Bomb rack unit.

C

C2W. Command and control warfare.

C3. Command, control and communications.

CAD. Cartridge activated device.

CAL. Confined area landing.

CASEVAC. Casualty evacuation.

CATCC. Carrier air traffic control center.

CATM. Captive air training missile.

CB. Circuit breaker.

CCA. Carrier controlled approach.

CCM. Counter-countermeasures.

CCR. Closed-circuit refueling.

CDU. Central display unit.

CFIT. Controlled flight into terrain.

cg. Center of gravity.

CHAL. Challenge.

CHAL CC. Challenge correct code.

CHAN. Channel.

CIC. Combat information center.

CIG. Missile cloud ceiling.

CM. Communication.

COMREL. Communications relay.

COMSEC. Communications security.

CONREP. Connected replenishment.

CPU. Central processing unit.

CRM. Crew resource management.

CRP. Contingency range power.

CRRC. Combat rubber raiding craft.

CSAR. Combat search and rescue.

CSCG. Communication system control group.

D

DA. Density altitude.

DAFCS. Digital automatic flight control system.

DAME. Distance azimuth measuring equipment.

DC. Direct current.

DFG. Directional finding group, ARA-50.

DECR. Decrease.

DECU. Digital electrical control unit.

DESU. Digital electronic sequence unit.

DLQ. Deck landing qualification.

DM. Decision making.

DME. Distance measuring equipment.

DNA. Designated Naval Aviator.

DSP. Droop stop pounding.

E

ECA. Electronic control amplifier.

ECS. Environmental control system.

ECU. Electrical control unit.

EHE. Estimated horizontal error.

ELT. Emergency locator transmitter.

ELVA. Emergency low visibility approach.

EMCON. Emissions control.

EMI. Electromagnetic interference.

EOD. Explosive ordnance disposal.

EOT. Element-on-time; Blade de-ice system.

ESM. Electronic support measures.

ESP. Enhanced survivability package.

ESU. Electronic sequence unit; APU.

ETF. Engine torque factor.

EW. Electronic warfare.

F

FCP. Functional Checkflight Pilot.

FD. Free deck.

FDD. Flight deck director.

FHS. FLIR hellfire system.

FLIR. Forward-looking infrared.

FMCP. Fuel management control panel.

FOD. Foreign object damage.

FOV. Field of view.

fpm. Feet per minute.

fps. Feet per second.

FRS. Fleet replacement squadron.

FSII. Fuel system icing inhibitor.

FTP. Fly-to-point.

G

GCA. Ground controlled approach.

GCU. Generator control unit.

GPS. Global positioning system.

GRP. Grid reference point.

H

H2P. Helicopter Second Pilot.

HABD. Helicopter air breathing device.

HAC. Helicopter aircraft commander.

HARS. Heading attitude reference system.

HATS. Hover attack torpedo shot.

HCU. Hand control unit.

HCO. Helicopter control officer.

HEELS. Helicopter emergency egress lighting system.

HF. High frequency.

HFS. Hellfire system.

HIFR. Helicopter in-flight refueling.

HIGE. Hover in ground effect.

HIT. Health indicator test.

HMU. Hydromechanical unit.

HOGE. Hover out of ground effect.

HRS. Horizon reference system.

HTW. Helicopter threat warning.

I

IAS. Indicated air speed.

IB. Interconnecting box.

ICS. Intercommunication system.

IDECU. Improved digital electrical control unit.

IFF. Identification friend or foe.

IFOBRL. In-flight operable bomb rack lock.

IFR. Instrument flight rules.

IGE. In ground effect.

IGV. Inlet guide vanes.

IHEELS. Individual helicopter emergency egress lighting system.

IMC. Instrument meteorological conditions.

INCR. Increase.

IPL. Initial program load.

IPS. Inlet particle separator.

IRCM. Infrared countermeasures.

IRP. Intermediate range power.

ISD. Integrated self-defense.

ISO. Instructor Sensor Operator.

ITO. Instrument takeoff.

J

JETT. Jettison.

JSO. Journeyman Sensor Operator.

K

KGS. Knots ground speed.

KIAS. Knots indicated airspeed.

L

LD. Leadership.

LDI. Leak detection/isolation.

LDS. Load demand spindle.

LEP. Laser eye protection.

LHEP. Left hand extended pylon.

LPU. Life preserver unit.

LRD. Laser range-finder and designator.

LSE. Landing signalman, enlisted.

LSO. Landing safety officer.

LTCH. Latch.

LTE. Loss of tail rotor effectiveness.

LVDT. Linear variable displacement transducer.

LZ. Landing zone.

M

MA. Mission analysis.

MAD. Magnetic anomaly detector.

MAF. Maintenance action form.

MAP. Missed approach point.

MAR. Mission avionics rack.

MARS. Mobile aircrew restraint system.

MEDEVAC. Medical evacuation.

MLA. Missle launcher assembly.

MLM. Marine location marker (smoke marker).

MPD. Multipurpose display.

MR. Main rotor.

MRC. Maintenance requirement card.

MSO. Master Sensor Operator.

MTMU. Magnetic tape memory unit.

MWS. Missile warning system.

N

NATIP. Naval Aviation Technical Information Product.

NATOPS. Naval Air Training and Operating Procedures Standardization.

NAVAIDS. Navigation aids.

N_g. Gas generator turbine speed.

NHC. NATO high capacity.

NI. NATOPS Instructor.

NOE. Nap of the earth.

NOTAM. Notice to airman.

N_p. Power turbine speed.

N_r. Main rotor speed.

NSFS. Naval surface fire support.

NSIU. Navigation switching interface unit.

NS-TL. Nose-tail.

NSV. Non synchronous vibrations.

NSW. Naval special warfare.

NTRP. Naval Technical Reference Procedures.

NTTP. Navy Tactics, Techniques, and Procedures.

NVD. Night vision device.

O

OAT. Outside air temperature.

ODV. Overspeed drain valve.

OEI. One engine inoperative.

OFT. Operational flight trainer.

OGE. Out of ground effect.

OLS. Optical landing system.

OOD. Officer of the deck.

ORIDE. Override.

ORM. Operational risk management.

ORT. Operational readiness test.

OTH. Over the horizon.

OTPI. On-top position indicator.

P

P_3. Compressor discharge pressure.

PA. Pressure altitude.

PAC. Pilot at the controls.

PAS. Power available spindle.

PBA. Pitch bias actuator.

PCL. Power control lever.

PCR. Pitch change rod.

PDB. Power distribution box.

PDI. Pressure differential indicator.

PDU. Pilot display unit.

PIC. Pilot in command.

PIM. Pulse interval modulation.

PIO. Pilot induced oscillations.

PNAC. Pilot not at the controls.

PPH. Pounds per hour.

PPS. Precise positioning system.

PQM. Pilot Qualified in Model.

PRF. Pulse repetition frequency.

PRI. Pulse repetition interval.

PTT. Push to talk.

R

RA. Recovery assist.

RADALT. Radar altimeter.

RAST. Recovery assist, secure and traverse system.

RAWS. Radar altimeter warning system.

RDP. Radar data processor.

RF. Radio frequency.

RLQ. RAST landing qualification.

RNS. Radar navigation system.

RSC. Remote switching console.

RSD. Rapid securing device.

R/T. Receiver/transmitter.

RTR. Rotor.

S

SA. Spectrum analyzer.

SAC. Standard airborne computer.

SAD. Safe/arm device.

SAR. Search and rescue.

SAS. Stability augmentation system.

SCA. Self-contained approach.

SDC. Signal data converter.

SDLM. Standard depot level maintenance.

SGSI. Stabilized glide slope indicator.

SIF. Selective identification feature.

SLC. Sonobuoy launch container.

SLEP. Service life extension program.

SO. Sensor operator.

SPIE. Special purpose insertion extraction.

SPS. Standard precisioning system.

SRT. Standard rate turn.

STR. Specification torque ratio.

SUS. Signal underwater sound.

SUW. Surface warfare.

T

T_2. Compressor inlet temperature.

TAS. True air speed.

TDI. Transient droop improvment.

TERF. Terrain flight.

TFO. Temporary flight orders.

TGT. Turbine gas temperature.

TGT REF. Turbine gas temperature reference.

TGW. Tail guide winch.

THP. Tape handling package.

TR. Torque ratio.

TRQ. Torque.

TRQ ADJ. Torque adjusted.

TRQ MEAS. Torque measured.

TRVS. Traverse.

TST. Test.

TTV. Target torque value.

TU. Turret unit (infrared laser receiver-transmitter-converter).

U

UL. Unlock.

USW. Undersea warfare.

V

VATS. Vibration analysis testing system.

VCR. Video cassette recorder.

Vd. Doppler-measured drift velocity, lateral.

VERTREP. Vertical replenishment.

VG. Variable geometry.

V_h. Doppler-measured velocity in the direction of aircraft heading, longitudinal.

VIDS. Vertical instrument display system.

VIDS/MAF. Visual discrepancy system/maintenance action form

VLA. Vertical launch assembly; visual landing aid.

VMC. Visual meteorological conditions.

VNT. Vent.

VSI. Vertical speed indicator.

VSR. Voltage sensor relay.

Vz. Doppler-measured vertical velocity.

W

WCA. Warning/caution/advisory.

WST. Weapon system trainer.

WPN. Weapon.

WPS. Waveguide pressurization system.

WOW. Weight-on-wheels.

X

XFD. Crossfeed.

PREFACE

SCOPE

This NATOPS manual is issued by the authority of the Chief of Naval Operations and under the direction of Commander, Naval Air Systems Command in conjunction with the Naval Air Training and Operating Procedures Standardization (NATOPS) program. It provides the best available operating instructions for most circumstances, but no manual is a substitute for sound judgment. Operational necessity may require modification of the procedures contained herein. Read this manual from cover to cover. It's your responsibility to have a complete knowledge of its contents.

APPLICABLE PUBLICATIONS

The following applicable publications complement this manual:

NTRP 3-22.2-SH60B (NATIP)

NTRP 3-22.4-SH60B (NATIP)

NTTP 3-22.5-ASW (ASW Tactical Pocket Guide)

NAVAIR A1-H60BB-NFM-500 (Pilot's Pocket Checklist)

NAVAIR A1-H60BB-NFM-700 (Functional Checkflight Checklist)

NAVAIR A1-H60BB-NFM-800 (Aircrew Pocket Checklist)

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The address of the Model Manager of this aircraft/publication is:

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Jacksonville, FL 32228-0118

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Revised text is indicated by a black vertical line in either margin of the page, like the one printed next to this paragraph. The change symbol shows where there has been a change. The change might be material added or information restated. A change symbol in the margin by the chapter number and title indicates a new or completely revised chapter.

WARNINGS, CAUTIONS, AND NOTES

The following definitions apply to WARNINGs, CAUTIONs, and Notes found throughout the manual.

WARNING

An operating procedure, practice, or condition, etc., that may result in injury or death, if not carefully observed or followed.

CAUTION

An operating procedure, practice, or condition, etc., that may result in damage to equipment, if not carefully observed or followed.

Note

An operating procedure, practice, or condition, etc., that is essential to emphasize.

WORDING

The concept of word usage and intended meaning adhered to in preparing this manual is as follows:

1. “Shall” has been used only when application of a procedure is mandatory.
2. “Should” has been used only when application of a procedure is recommended.
3. “May” and “need not” have been used only when application of a procedure is optional.
4. “Will” has been used only to indicate futurity, never to indicate any degree of requirement for application of a procedure.
5. “Land immediately” is self-explanatory.
6. “Land as soon as possible” means land at the first site at which a safe landing can be made.
7. “Land as soon as practical” means extended flight is not recommended. The landing and duration of flight is at the discretion of the pilot in command.

Note

This manual shall be carried in the aircraft at all times.

CHANGE RECOMMENDATIONS

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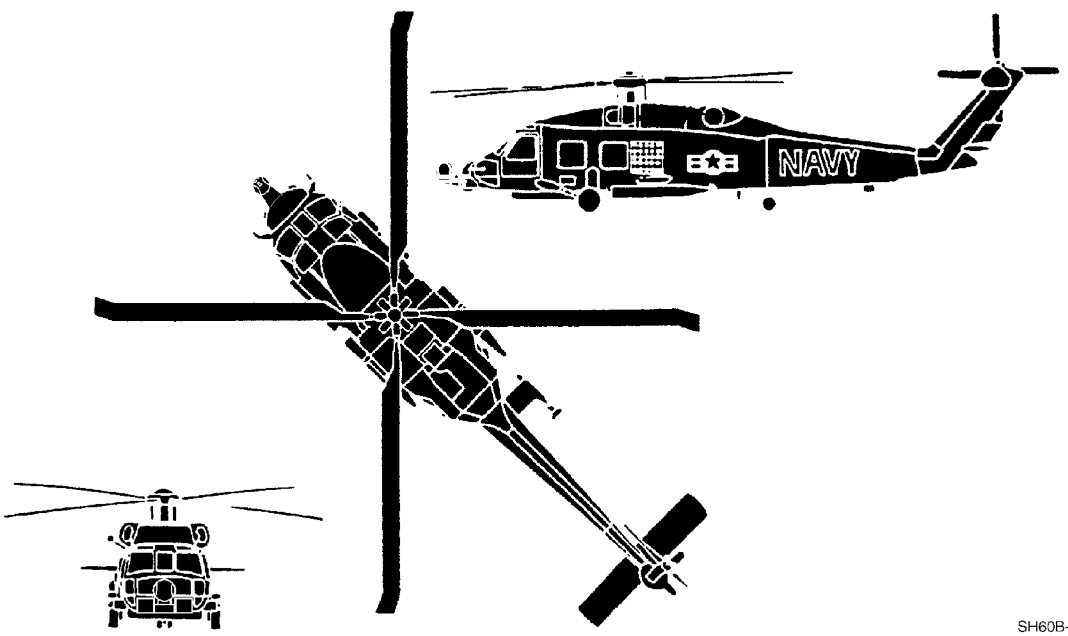


SH60B-F003

PART I

The Aircraft

CHAPTER 1

General Description

1.1 THE HELICOPTER

The SH-60B (SEAHAWK) is a single main rotor, twin-engine helicopter, manufactured by United Technologies Corporation, Sikorsky Aircraft Division. The helicopter has a 20° tractor type canted tail rotor, a controllable stabilator, conventional fixed landing gear, an external cargo hook, a rescue hoist, and bomb racks for carrying and launching external stores. In addition, it is equipped with a flight-rated auxiliary power unit, a sonobuoy launch system, an anti-ice system, a fire-extinguishing system, an environmental control system, an automatic flight control system (AFCS), a single-point pressure refueling system, a helicopter in-flight refueling (HIFR) system, and the necessary avionics and instrumentation for instrument flight and mission accomplishment. The helicopter design is compatible with ships equipped with a recovery, assist, securing and traversing (RAST) system, and the main rotor blades and tail pylon can be folded for storage. In addition, the helicopter can operate from non-RAST equipped combatants and a variety of other naval ships.

1.2 DIMENSIONS

The overall aircraft dimensions and clearances (Figure 1-1 and Figure 1-2) are:

Folded length (rotor/tail pylon) . . . . . 40 ft, 11 in

Folded length (rotor/tail pylon) (FHS equipped) . . . . . 42 ft, 10 in

Rotor folded length (pylon flight position) . . . . . 53 ft, 3 in

Length overall (rotors turning) . . . . . 64 ft, 10 in

Fuselage length . . . . . 50 ft, 0 in

Height . . . . . 17 ft, 0 in

Fuselage width . . . . . 7 ft, 9 in

Folded width . . . . . 10 ft, 7 in

Unfolded width (FHS equipped) . . . . . 13 ft, 0 in

Main rotor diameter . . . . . 53 ft, 8 in

Tail rotor diameter . . . . . 11 ft, 0 in

Ground clearance . . . . . 11.2 in

Ground clearance (FHS equipped) . . . . . 11 in

Ground to tail rotor clearance . . . . . 6 ft, 8 in

Turning radius . . . . . 41 ft, 7 in

Clearance for 180° . . . . . 84 ft, 0 in

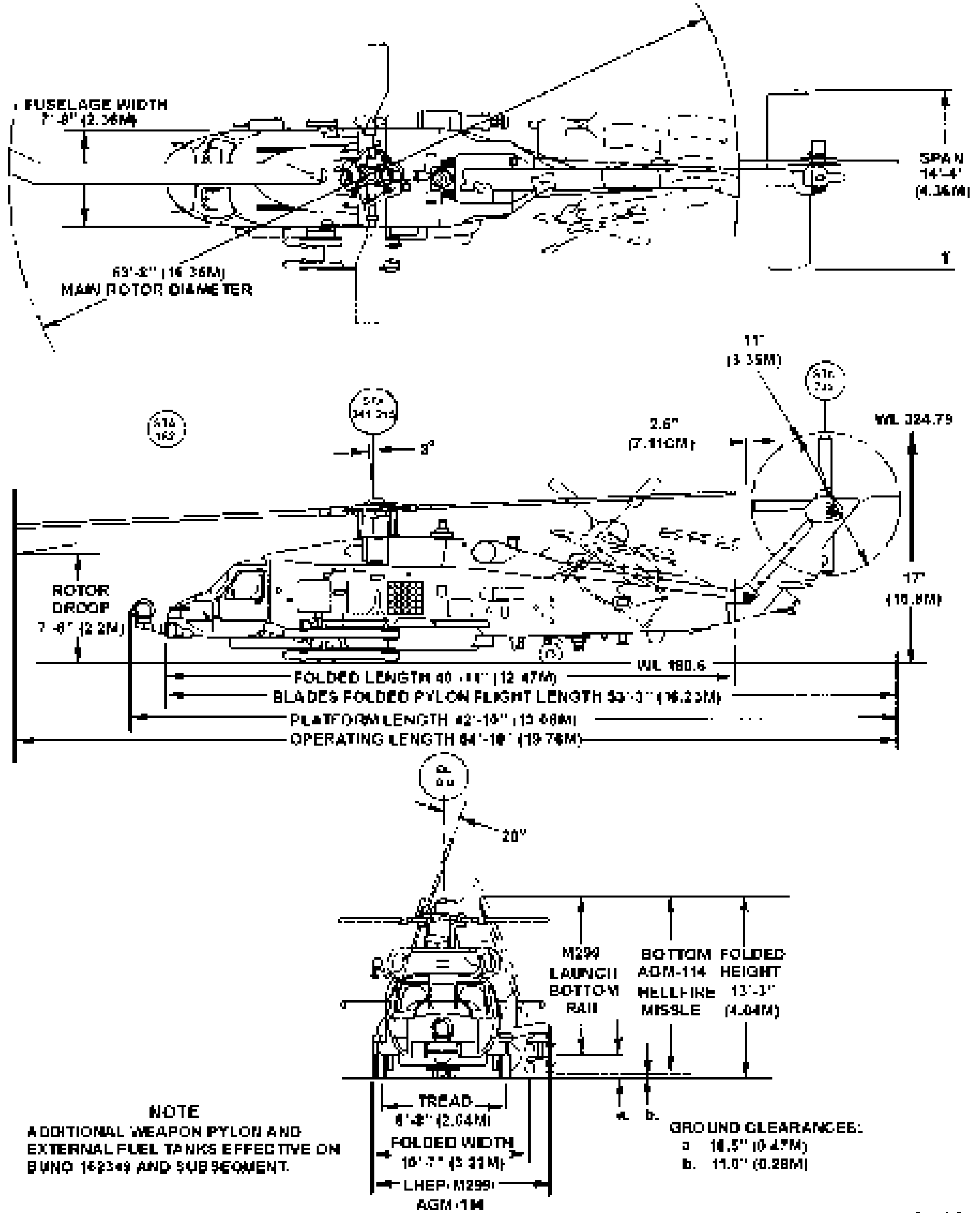


Figure 1-1. Principal Dimensions

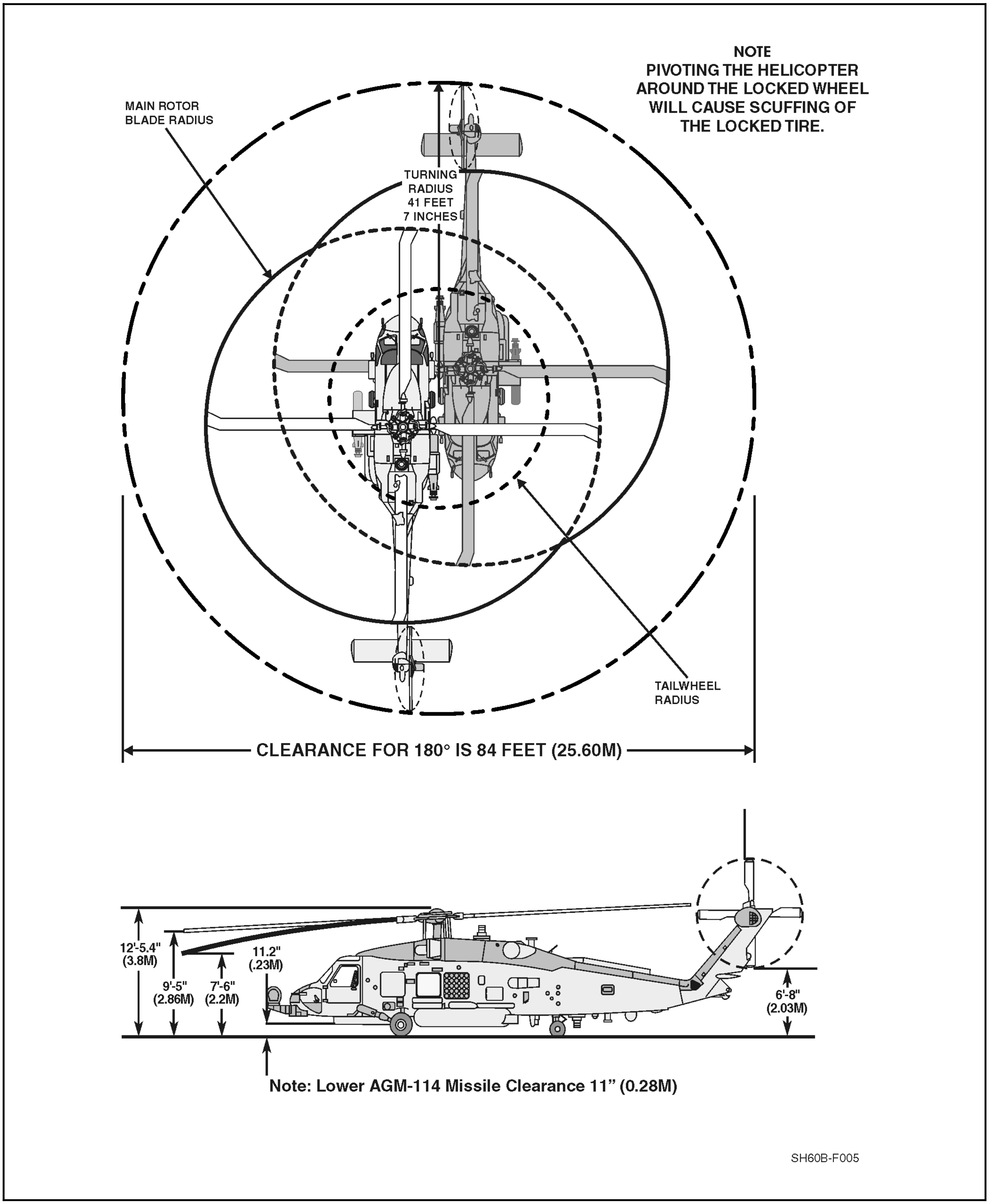


Figure 1-2. Ground Clearance and Turning Radius

1.3 THE ENGINES

The helicopter is equipped with two T700–GE–401C engines. The T700–GE–401C (Figure 2-1) is a front–drive turboshaft engine, manufactured by the General Electric Company, Aircraft Engine Group. Some of the features of the engine include an integral inlet particle separator and self–contained systems incorporating modular construction. At sea level and 59 °F (15 °C), the T700–GE–401C shaft horsepower ratings are:

Contingency: 2 1/2 Minute Duration 1,940

Intermediate: 30 Minute Duration 1,800

Max Continuous: No limit 1,662

1.4 GENERAL ARRANGEMENT

The following subsections describe the exterior and interior arrangements of the aircraft.

1.4.1 Exterior Arrangement

The exterior arrangement of the aircraft is shown in Figure 1-3. Also shown are the locations of the aircraft antennas. There are two data link antennas: one forward and one aft on the underside of the aircraft. The search radar antenna is also located on the underside of the aircraft. Other antennas (UHF/VHF, HF, radar altimeter, TACAN, ESM, sonobuoy receivers, Doppler, ADF, IFF, and GPS) are located at various points on the helicopter (Figure 1-3).

The left inboard, left outboard, and right weapon pylons accommodate BRU–14/A weapon/stores racks. Fittings for torpedo parachute release lanyards are located on the fuselage aft of each weapon pylon. Effective on BuNo 162349 and subsequent, the left and right inboard pylons have wiring and tubing provisions for auxiliary fuel tanks. All pylons have wiring provisions to accommodate the Mk 50 torpedo. A FLIR turret may be mounted on the right hand pylon or nose mounted bracket. The left outboard weapon pylon can accommodate an M299 Launcher and up to four AGM–114 Hellfire missiles, or M36E1 CATMs.

The magnetic anomaly detector (MAD) towed body and reeling machine are mounted on a faired structure that extends from the forward tail–cone transition section on the right side of the aircraft. It is positioned above and aft of the right weapon pylon.

The sonobuoy launcher is located on the left side of the aircraft, above the left weapon pylon. The sonobuoy launcher is loaded from ground level outside the aircraft. Sonobuoys are pneumatically launched laterally to the left of the aircraft.

The airborne RAST system main probe and external cargo hook are on the bottom fuselage centerline, just aft of the main rotor centerline.

Fuel service connections, for both gravity and pressure refueling, are located on the left side of the aircraft, aft of the weapon pylons. Dual–engine waterwash is manifolded from a single–point selector valve connector on the left side of the aircraft above the window of the sensor operator (SO). A connector to service the sonobuoy launcher nitrogen bottle is located next to the waterwash connector.

The long strokes of both main and tail wheel oleos are designed to dissipate high–sink–rate landing energy. Axle and high–point tiedowns are provided at each main gear. Fuselage attachments are provided above the tail gear for connection to the RAST tail–guide winch system allowing aircraft maneuvering and straightening aboard ship and for tail pylon tiedown.

1.4.2 Interior Arrangement

Figure 1-4 shows the cockpit and cabin arrangement. Hinged doors on each side of the cockpit provide normal access to and from that station. A sliding door on the right side of the fuselage provides access to and from the cabin. The primary emergency escape routes are:

1. Pilot, right-hand jettisonable window.
2. ATO, left-hand jettisonable window.
3. SO, left-hand jettisonable window.
4. Instructor/passenger, jettisonable cabin door window.

The SO console (Figure 1-4) is located in the cabin, as well as provisions for a removable instructor/passenger seat, a passenger seat, and a litter.

Figure 1-5 shows a diagram of the pilot compartment. The ATO station is located on the left side of the aircraft cockpit. It is equipped with, or offers access to, a full complement of aircraft flight controls and instruments. Figure 1-8 illustrates the pilot and ATO instrument panel.

The overhead console (Figure 1-6), located above the pilot and ATO stations, contains aircraft system control panels involving circuit breakers, console/instrument light controls, external light controls, fire-extinguisher controls, engine controls, and several miscellaneous controls. The lower console (Figure 1-7) is located in the cockpit between the pilot and ATO stations. It contains the ATO avionics, AFCS, and communications controls. The lower console is accessible by either the ATO or the pilot.

The ATO keyset (Figure 1-7) is located on the lower console. The multipurpose display (MPD) is located on the instrument panel between the ATO flight instrument panel and a caution/advisory panel (Figure 1-8).

The collective on the ATO side telescopes to allow improved cockpit ingress and egress. In addition, locations are provided in the cabin for two fire extinguishers, two first aid kits, two canteens, a crash ax, a map case, and a backup messenger kit.

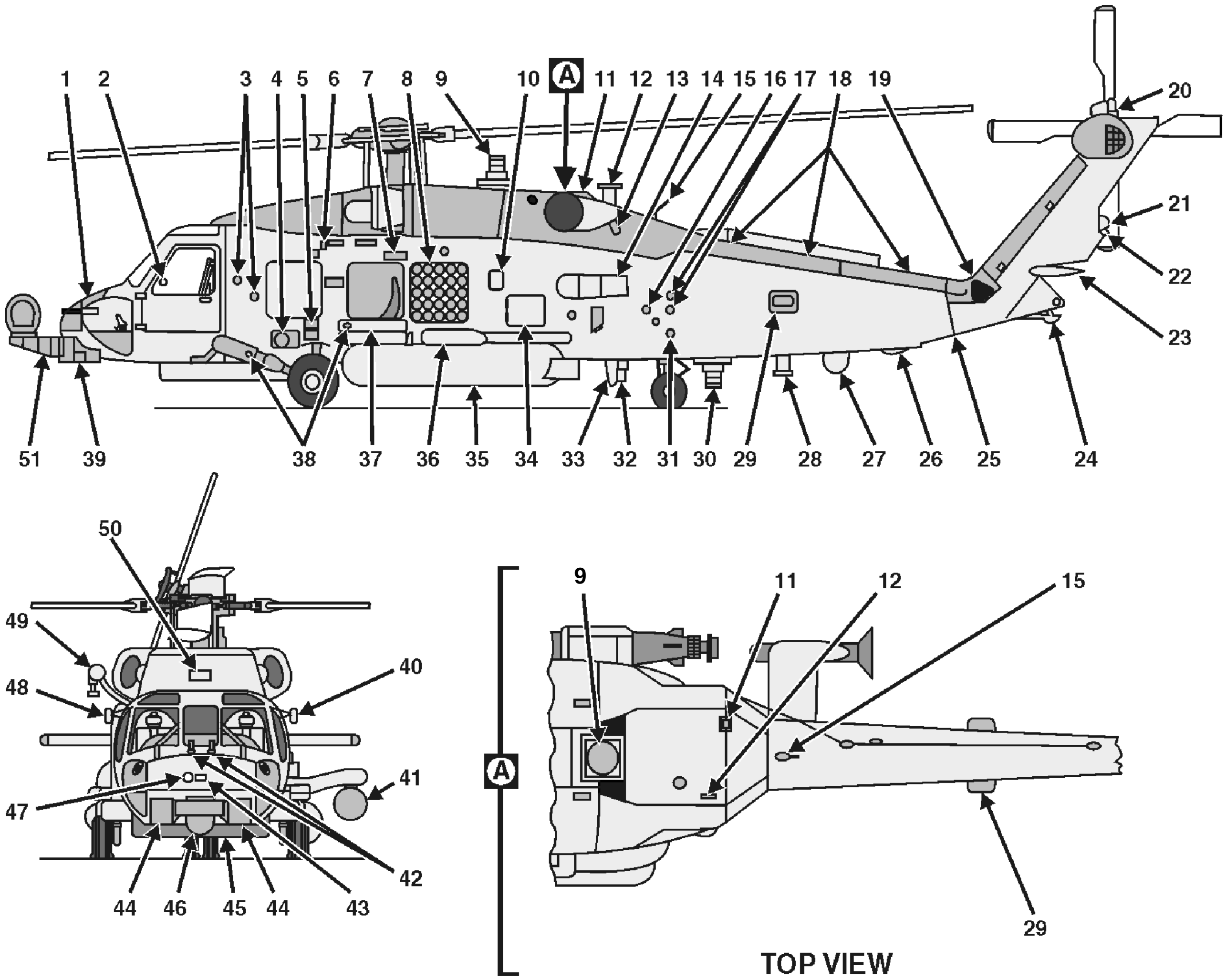
The cabin is arranged with the SO station on the left, facing forward (Figure 1-4). Most of the components of the avionics system are physically located in the SO console rack, situated aft of the ATO seat, and in the mission avionics rack (MAR), situated aft of the pilot seat. Figure 1-9 shows the location of the individual panels on the SO console and the SO keyset. The SO console contains the necessary controls and indicators for the SO to perform the missions of surface warfare (SUW) and undersea warfare (USW).

To the right of the SO station seat is a seat which accommodates an instructor or a passenger. The primary passenger seat is in the instructor seat position during single passenger flights. The hoist controls and hover-trim panel are located adjacent to the cabin door. The cargo hook hatch is located forward of the RAST probe housing.

1.5 MISSION OVERVIEW

LAMPS is the acronym for light airborne multipurpose system. The SH-60B helicopter is configured specifically in response to the LAMPS requirement of the U.S. Navy. The LAMPS MK III system has been designed to the Navy sea control mission. In fulfilling the mission, LAMPS MK III will encounter a threat that has many dimensions. The threat encompasses a hostile submarine fleet and missile-equipped surface ships. The system extends the search and attack capabilities of LAMPS MK III configured destroyer, frigate, and cruiser platforms, deploying helicopters directly from these ships.

The primary missions of the LAMPS MK III are those of SUW and USW. Aircraft prior to BuNo 162349 are capable of the antiship surveillance and targeting (ASST) and USW roles only. Effective with BuNo 162349 and subsequent, LAMPS MK III Block I aircraft equipped with the FLIR Hellfire system can be used in traditional SUW attack roles. Secondary missions include search and rescue (SAR), medical evacuation (MEDEVAC), vertical replenishment (VERTREP), naval surface fire support (NSFS), and communications relay (COMREL). Each of the primary and secondary missions will be discussed in the following subsections.



1. LEFT PITOT TUBE
2. COCKPIT VENT WINDOW
3. STATIC PORTS
4. AVIONICS COOLING EXHAUST
5. EXTERNAL ICS/ARM PANEL
6. MOORING SHACKLE
7. ENGINE WASH/SONOBUOY LAUNCHER CHARGER PANEL
8. SONOBUOY LAUNCHER
9. UPPER IRCM TRANSMITTER
10. GRAVITY REFUELING PORT
11. GPS ANTENNA
12. UPPER VHF/UHF/IFF ANTENNA
13. APU EXHAUST
14. ESM ANTENNA
15. EMERGENCY LOCATOR ANTENNA
16. FIRE EXTINGUISHER THERMAL PLUG
17. TAIL WHEEL STRUT INSPECTION WINDOW
18. TAIL ROTOR DRIVE SHAFT COVER
19. INTERMEDIATE GEAR BOX
20. UPPER ANTICOLLISION LIGHT
21. AFT MISSILE DETECTION SENSOR
22. TAIL POSITION LIGHT
23. STABILATOR
24. TAIL BUMPER
25. PYLON FOLD ASSIST CONNECTION
26. REMOTE COMPASS
27. AFT DATA LINK ANTENNA
28. LOWER VHF/UHF/TACAN ANTENNA
29. LEFT COUNTERMEASURES DISPENSER
30. LOWER IRCM TRANSMITTER
31. MOORING SHACKLE
32. LOWER ANTICOLLISION LIGHT
33. SONOBUOY ANTENNA
34. PRESSURE REFUELING PORT
35. EXTERNAL FUEL TANK
36. INBOARD LEFT PYLON
37. OUTBOARD LEFT PYLON
38. POSITION LIGHT
39. FORWARD MISSILE DETECTION SENSOR
40. REAR VIEW MIRROR
41. TORPEDO
42. OUTSIDE AIR TEMPERATURE SENSORS (STBD - DE-ICE, PORT - AOP)
43. AVIONICS COMPARTMENT COOLING DAMPER DOOR
44. FORWARD ESM ANTENNA
45. SEARCH RADAR ANTENNA
46. FORWARD DATA LINK ANTENNA
47. TACAN ANTENNA
48. PILOT'S REAR VIEW MIRROR
49. RESCUE HOIST
50. HYDRAULIC BAY COOLING INLET
51. NOSE MOUNT PLATFORM

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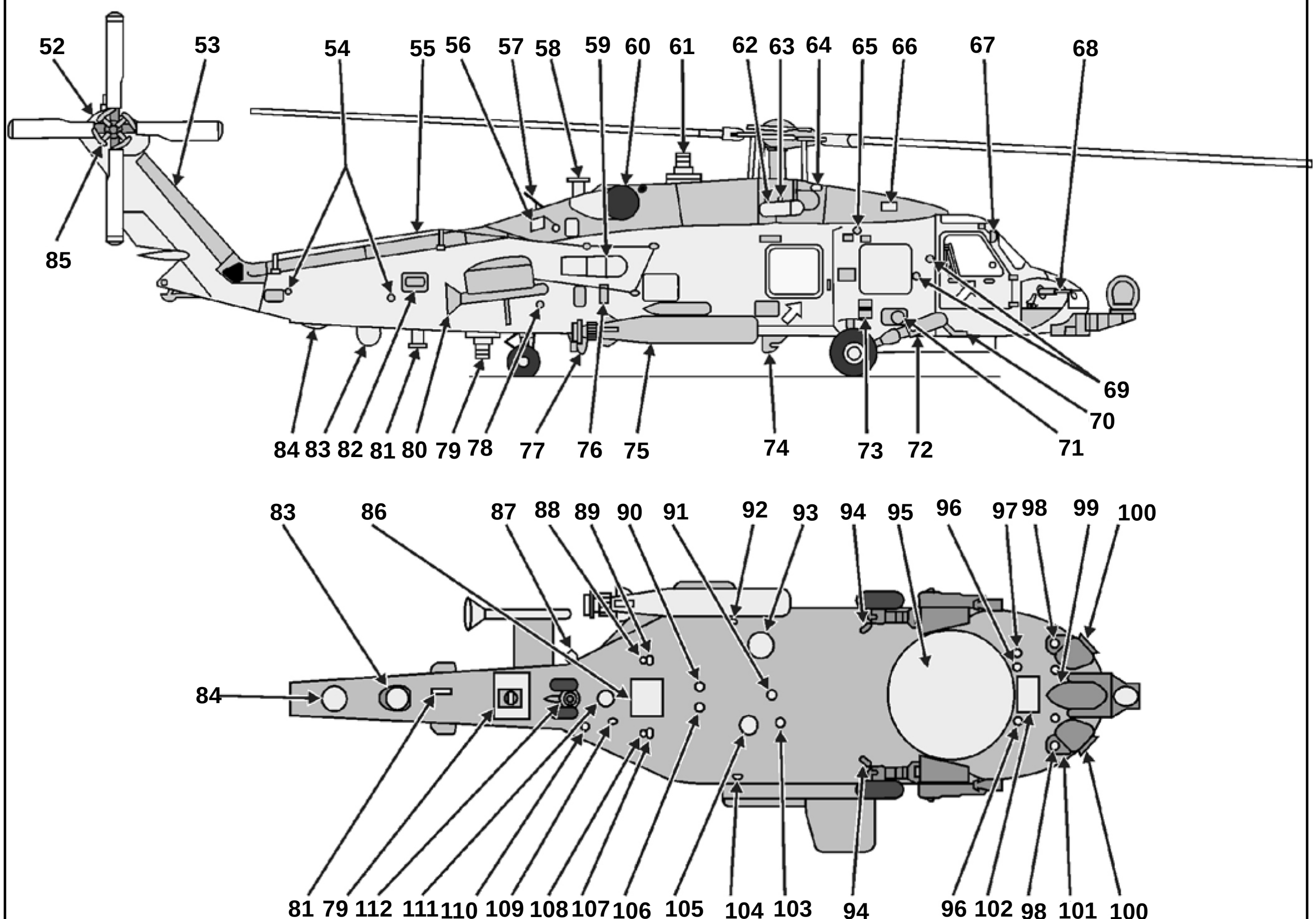
Figure 1-3. General Arrangement (Sheet 1 of 2)



52. TAIL ROTOR
53. PYLON DRIVE SHAFT COVER
54. BLADE STOWAGE SUPPORT
55. HF ANTENNA
56. ECS EXHAUST
57. EMERGENCY LOCATOR ANTENNA
58. UPPER VHF/UHF/IFF ANTENNA
59. ESM ANTENNA
60. GPS ANTENNA
61. UPPER IRCM TRANSMITTER
62. RESCUE HOIST
63. ICE DETECTOR
64. ROTOR HEAD LIGHT
65. MOORING SHACKLE
66. AVIONICS COOLING INTAKE
67. REAR VIEW MIRROR
68. RIGHT PITOT TUBE
69. STATIC PORT
70. COCKPIT STEP (BOTH SIDES)
71. AVIONICS COOLING EXHAUST
72. POSITION LIGHT
73. EXTERNAL POWER CONNECTOR
74. CARGO HOOK
75. TORPEDO (BOTH SIDES)
76. ECS AIR INLET
77. SONOBUOY ANTENNA
78. MOORING SHACKLE
79. LOWER IRCM TRANSMITTER
80. MAD AND REELING MACHINE
81. LOWER VHF/UHF/TACAN ANTENNA
82. RIGHT COUNTERMEASURES DISPENSER
83. AFT DATA LINK ANTENNA
84. REMOTE COMPASS
85. TAIL GEAR BOX
86. DOPPER ANTENNA
87. FUEL DUMP PORT
88. NO. 2 FUEL CELL VENT
89. NO. 2 ENGINE OVERBOARD DRAIN
90. RIGHT FUEL TANK SUMP DRAIN
91. MAIN RAST PROBE
92. TRANSMISSION DRIP PAN DRAIN
93. ADF/OTPI ANTENNA
94. HYDRAULIC SYSTEM OVERBOARD DRAIN
95. SEARCH RADAR ANTENNA
96. LAND/HOVER LIGHT
97. SEARCH LIGHT
98. RADAR ALTIMETER
99. FORWARD DATA LINK ANTENNA
100. FORWARD ESM ANTENNA
101. FORWARD MISSILE DETECTION SENSOR
102. FLOTATION BOTTLE ACCESS PANEL
103. MAIN RAST PROBE LIGHT
104. TRANSMISSION DRIP PAN DRAIN
105. LOWER IFF ANTENNA
106. LEFT FUEL TANK SUMP DRAIN
107. NO. 1 ENGINE AND APU OVERBOARD DRAIN
108. NO. 1 FUEL TANK VENT
109. SONOBUOY RECEIVING ANTENNA
110. TAIL PROBE LIGHT
111. LOWER ANTICOLLISION LIGHT
112. TAIL PROBE

SH60B-F007

Figure 1-3. General Arrangement (Sheet 2)

1.5.1 Primary Missions

In a USW mission, the aircraft is deployed from the parent ship to classify, localize, and possibly attack a suspected threat that has been detected by the shipboard towed-array sonar, hull-mounted sonar, or by other internal or external sources.

When used in an SUW mission, the aircraft provides a mobile, elevated platform for observing, identifying, and localizing threat platforms beyond the parent shipboard radar and/or electronic support measure (ESM) horizon. When a suspected threat is detected, classification and targeting data are provided to the parent ship via the data link for surface-to-surface weapon engagement. Hellfire missile equipped aircraft may conduct independent or coordinated attack, dependent upon the threat and tactical scenario. Refer to NTRP 3-22.4-SH60B for further explanation of USW and SUW missions.

1.5.2 Secondary Missions

In the VERTREP mission, the aircraft is able to transfer material and personnel between ships, or between ship and shore. Refer to Chapter 9.

In the SAR mission, the aircraft is designed to search for and locate a particular target/object/ship or plane and to rescue personnel using the rescue hoist. Refer to SAR Manual (NWP 3-50 Series), JCS 3-50, SAR TACAID A1-SARBA-TAC-000. Refer to Chapter 9.

In the MEDEVAC mission, the aircraft provides for the medical evacuation of ambulatory and litterbound patients.

In the COMREL mission, the aircraft serves as a receiver and transmitter relay station for over-the-horizon (OTH) communications between units. Refer to Chapter 15.

In the NSFS mission, the aircraft provides a platform for spotting and controlling naval gunfire from either the parent ship or other units. Refer to NTTP 3-22.5-ASW.

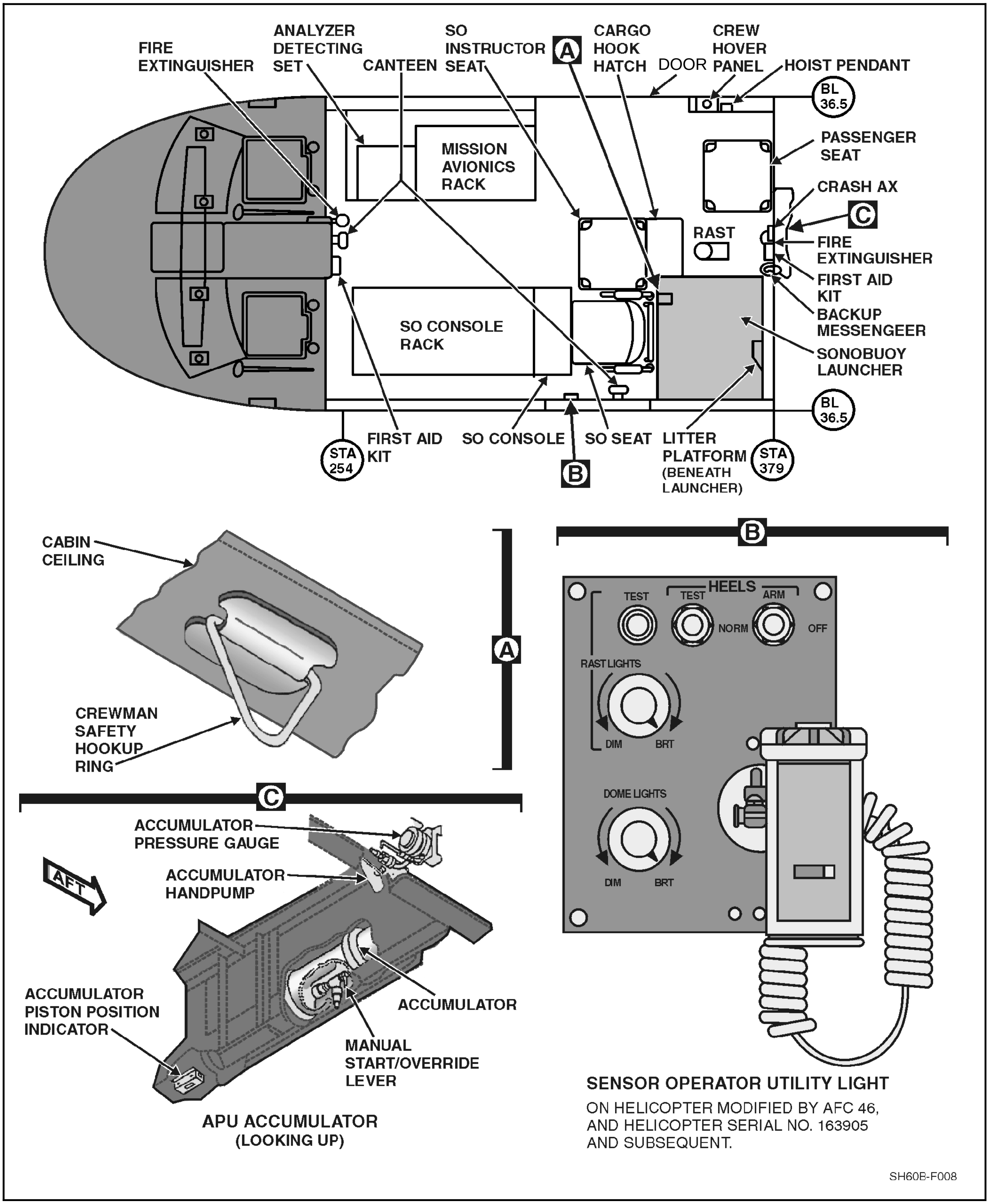
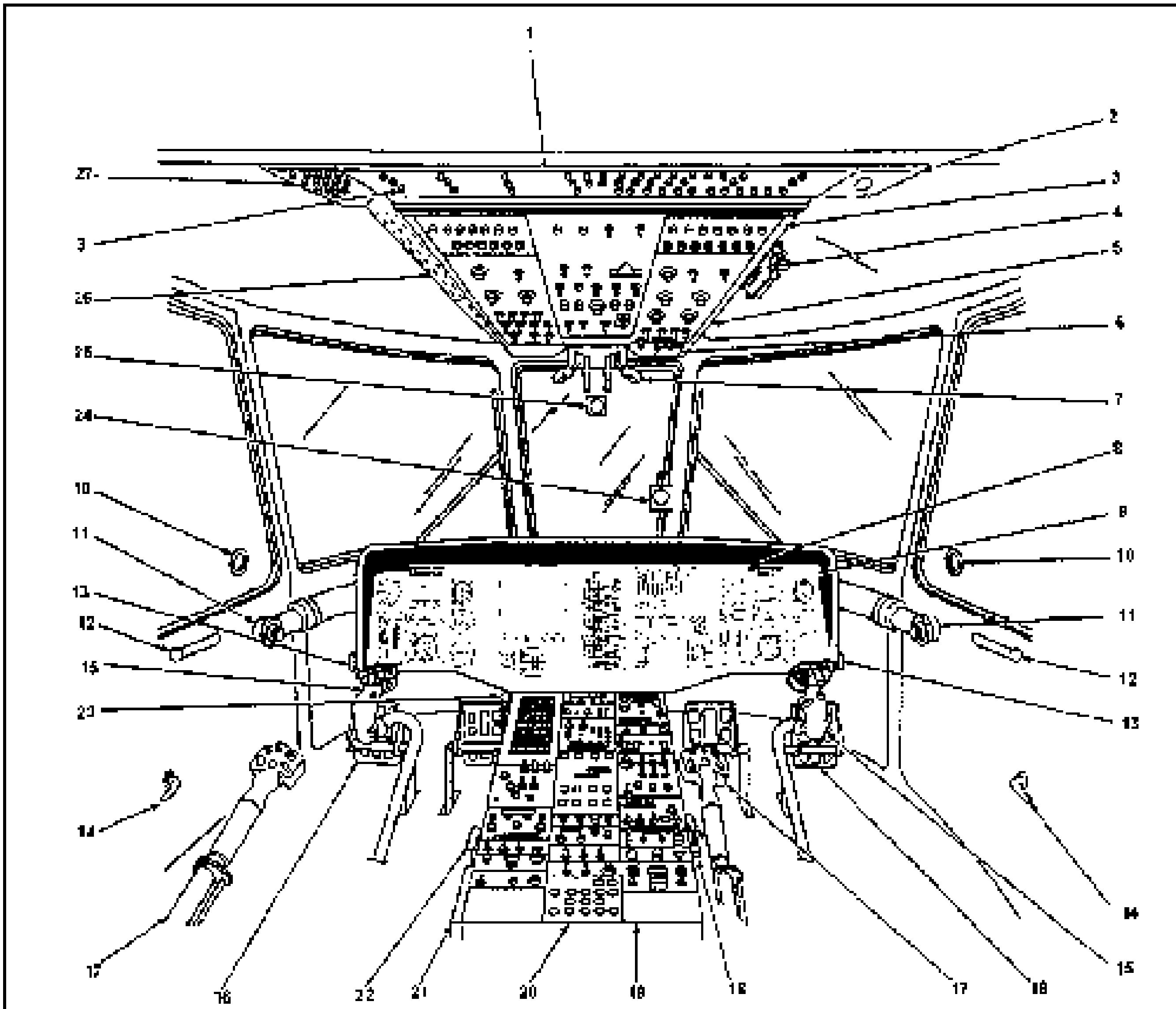


Figure 1-4. Cabin Arrangement



1. CENTER CIRCUIT BREAKER PANEL
2. COCKPIT SECONDARY/CONSOLE BREAKER PANEL
3. DC ESNTL BUS CIRCUIT BREAKER PANEL
4. ROTOR BRAKE MASTER CYLINDER
5. UPPER CONSOLE
6. STANDBY COMPASS CORRECTION CARD
7. ENGINE CONTROL QUADRANT
8. THREAT INDICATOR
9. INSTRUMENT PANEL
10. POPOUT AIR VENT
11. AIRDUCT
12. WINDOW EMERGENCY RELEASE HANDLE
13. PEDAL ADJUST HANDLE
14. INTERIOR DOOR HANDLE
15. CYCLIC GRIP
16. TAIL ROTOR PEDAL (TYPICAL)
17. COLLECTIVE GRIP (TYPICAL)
18. PARKING BRAKE HANDLE/TAIL WHEEL LOCK SWITCH
19. LOWER CONSOLE
20. BATT UTIL BUS CIRCUIT BREAKER PANEL
21. MAP CASE
22. RAST EMERGENCY RELEASE HANDLE
23. NOSE DOOR COOLING DAMPER CONTROL
24. STANDBY COMPASS
25. FREE-AIR TEMPERATURE INDICATOR
26. ATO CIRCUIT BREAKER PANEL
27. CORNER CIRCUIT BREAKER PANEL

Figure 1-5. Pilot Compartment

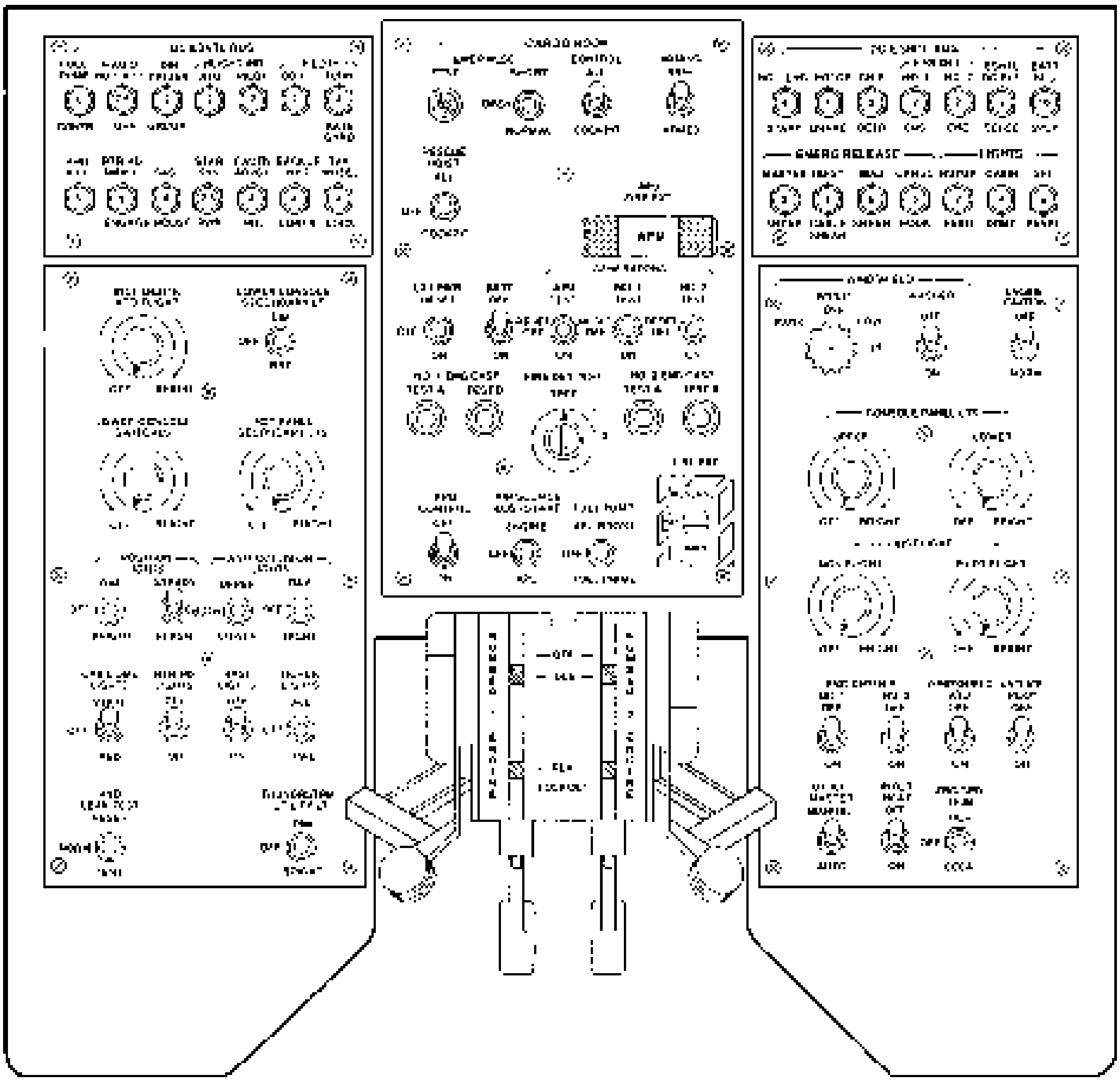


Figure 1-6. Overhead Console

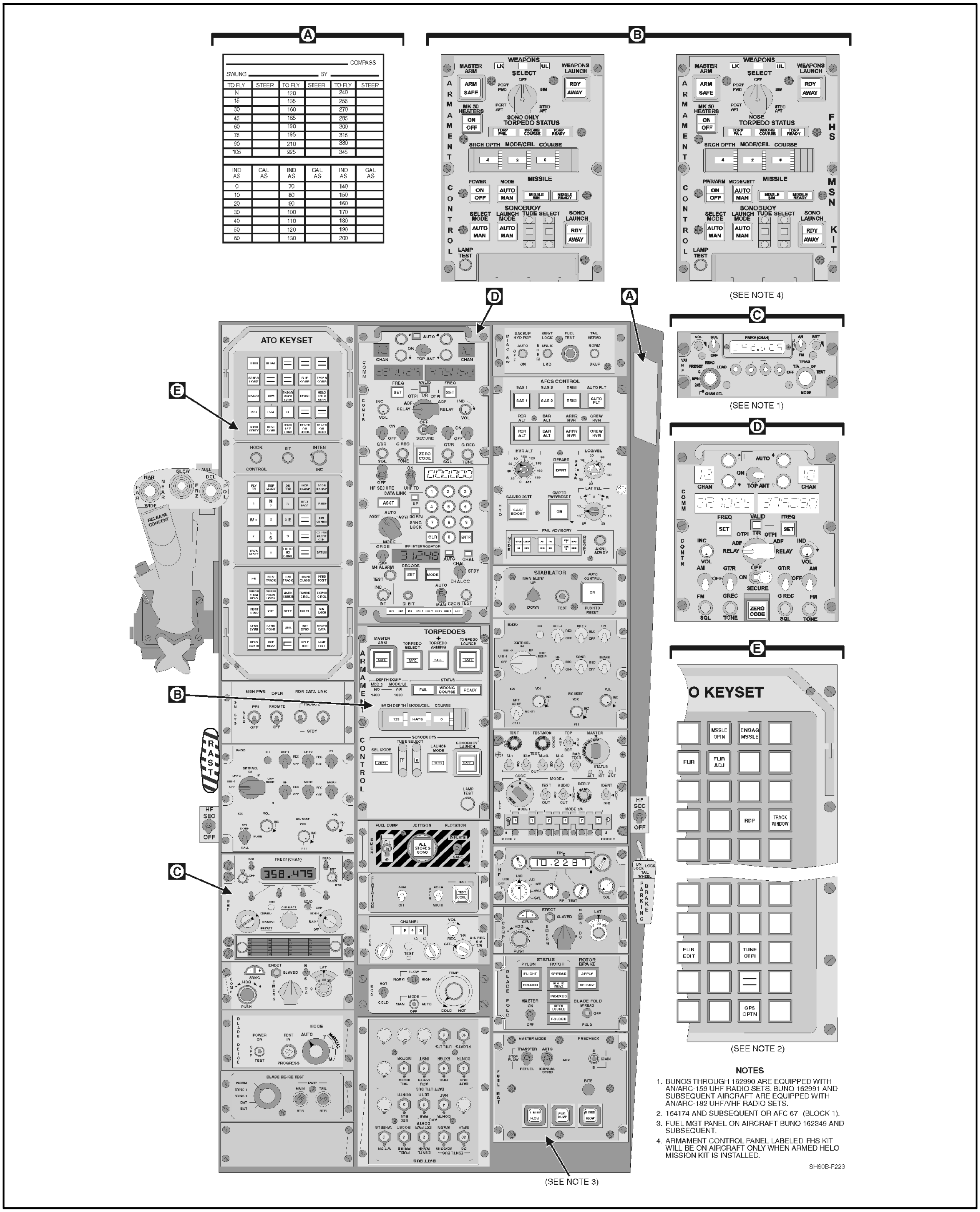


Figure 1-7. Lower Console

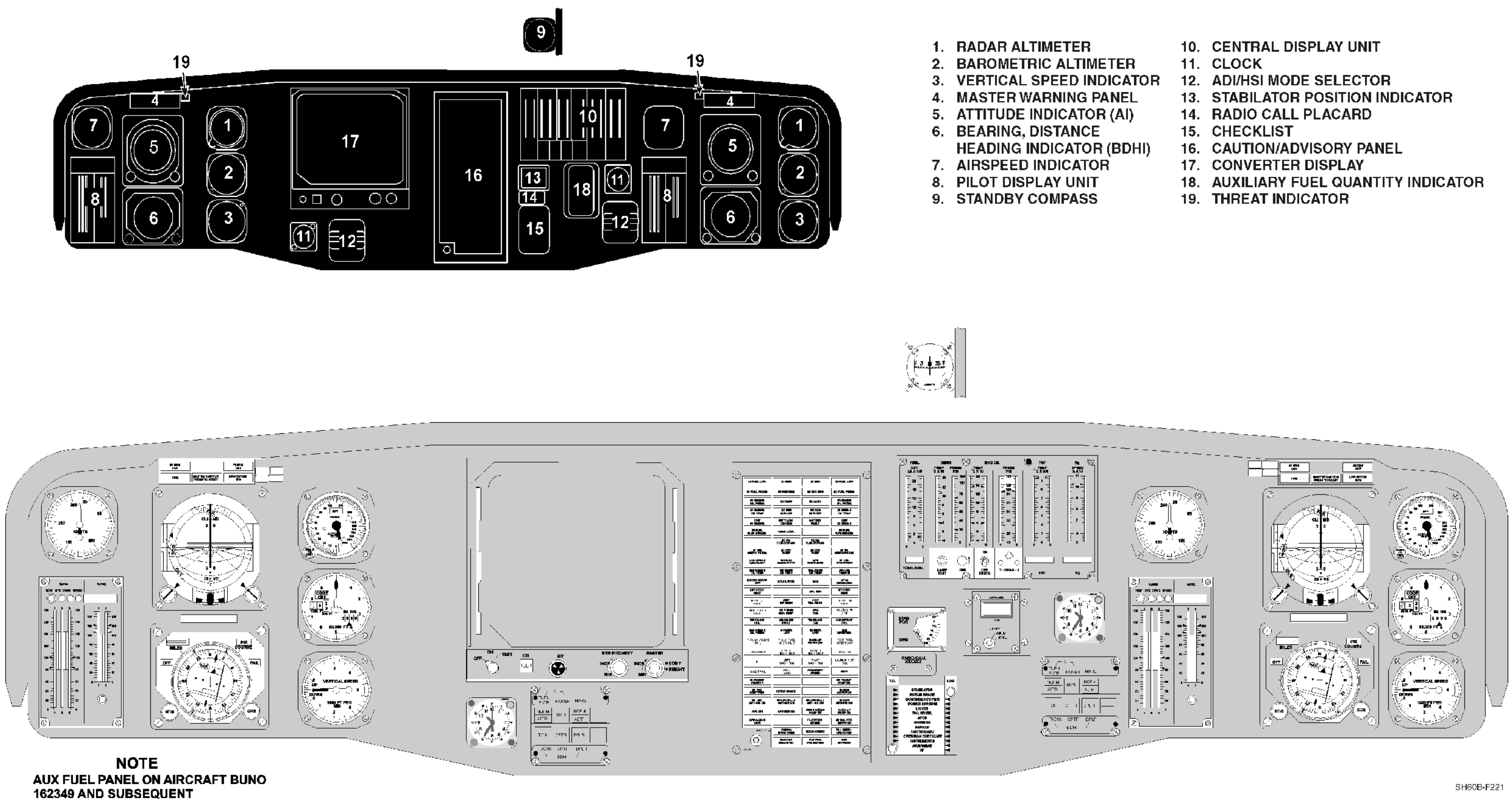


Figure 1-8. Instrument Panel

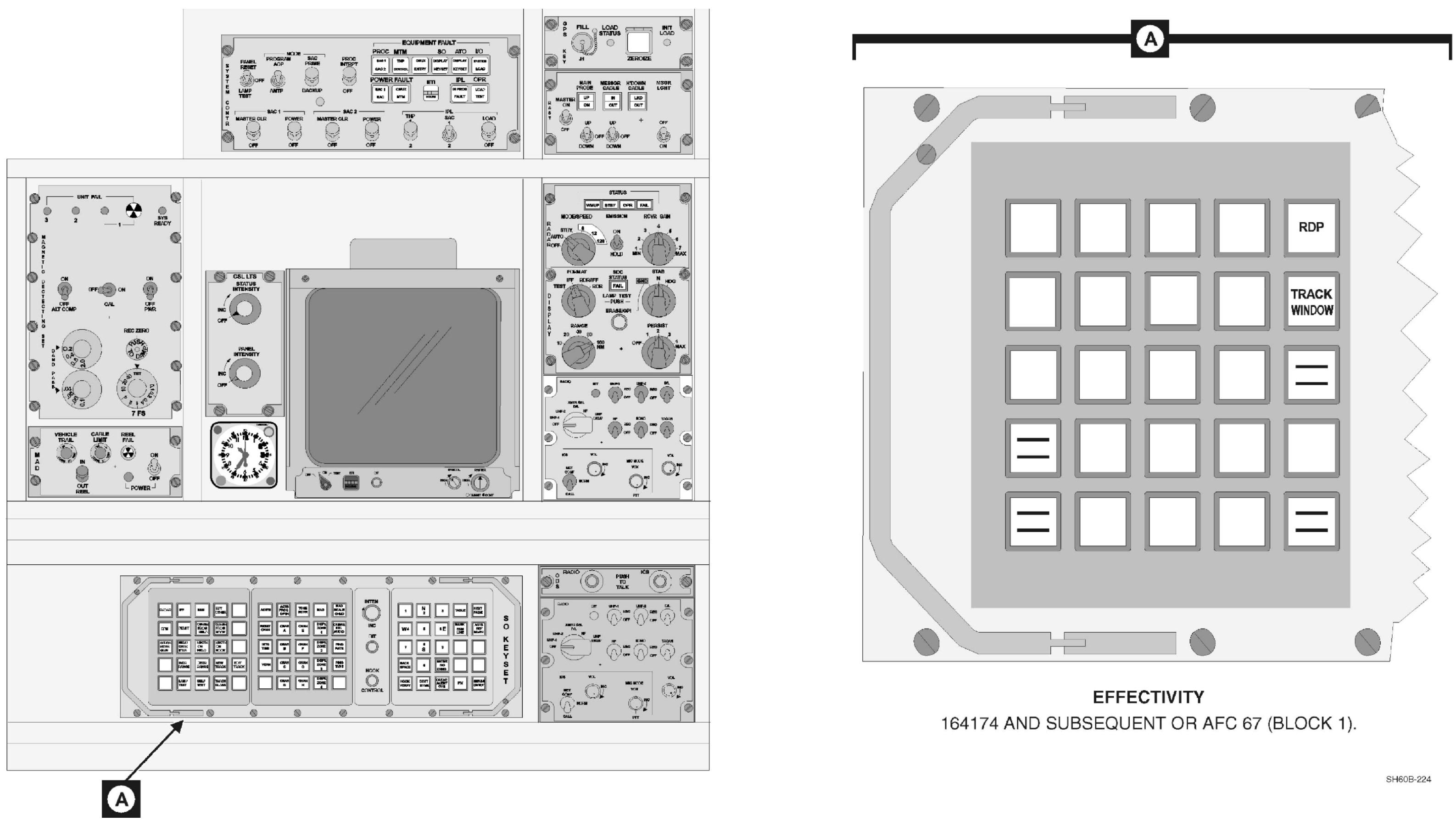


Figure 1-9. SO Console

CHAPTER 2

Systems

2.1 POWERPLANT SYSTEMS AND RELATED SYSTEMS

2.1.1 Engine

The T700-GE-401C engine (Figure 2-1) is a front-drive, turboshaft engine. It consists of five sections: inlet, compressor, combustor, turbine, and exhaust. The engine incorporates an integral waterwash manifold, inlet particle separator (IPS), a top-mounted accessory section, an engine-driven fuel boost pump, a hydro-mechanical control unit (HMU), an improved digital electronic control unit (DECU), a self-contained lubrication system, engine condition monitoring and diagnostics provisions.

2.1.1.1 Inlet Section

The inlet section includes the inlet cowling, swirl frame, front frame, main frame, accessory section, and scroll case. The waterwash manifold is an integral part of the swirl frame, which aims a series of jets into the compressor inlet area.

2.1.1.2 Inlet Air Flow

Ambient air first passes through the inlet cowling, the inside of which can be heated with fifth stage bleed air via the inlet anti-ice valve, to prevent ice from forming on the engine inlet. The inlet fairing is also perforated with small slits, to allow fifth stage bleed air to heat the exposed inlet surface for anti-icing purposes. Next, the intake airflow passes through the swirl vanes, which impart rotation to the airflow. The swirl vanes are hollow to permit passage of fifth stage bleed air via the engine anti-ice valve for anti-icing purposes.

2.1.1.3 Inlet Particle Separator

The Inlet Particle Separator (IPS) (Figure 2-2) prevents foreign particles from entering the compressor. The rotation of the airflow, imparted by the swirl vanes, causes the particles to be thrown outward into the collection scroll. Prior to being dumped overboard through the blower assembly, the IPS discharge airflow passes over the scroll vanes, which are contained in the collection scroll.

Clean air, free of foreign particles, then passes through the de-swirl vanes, which remove the rotation imparted by the IPS, providing smooth airflow to the compressor.

2.1.1.4 Compressor Section

The compressor section consists of a five-stage axial and a single-stage centrifugal rotor/stator assembly.

2.1.1.5 Combustion Section

The combustion section consists of a flow-through, annular combustion chamber, two igniters, and 12 fuel injectors that receive fuel via the Overspeed and Drain Valve (ODV), supplying atomized fuel for combustion.

2.1.1.6 Gas-Generator Turbine

The N_g turbine drives the compressor and accessory gearbox (AGB). It is a two-stage, air-cooled, high-performance axial design.

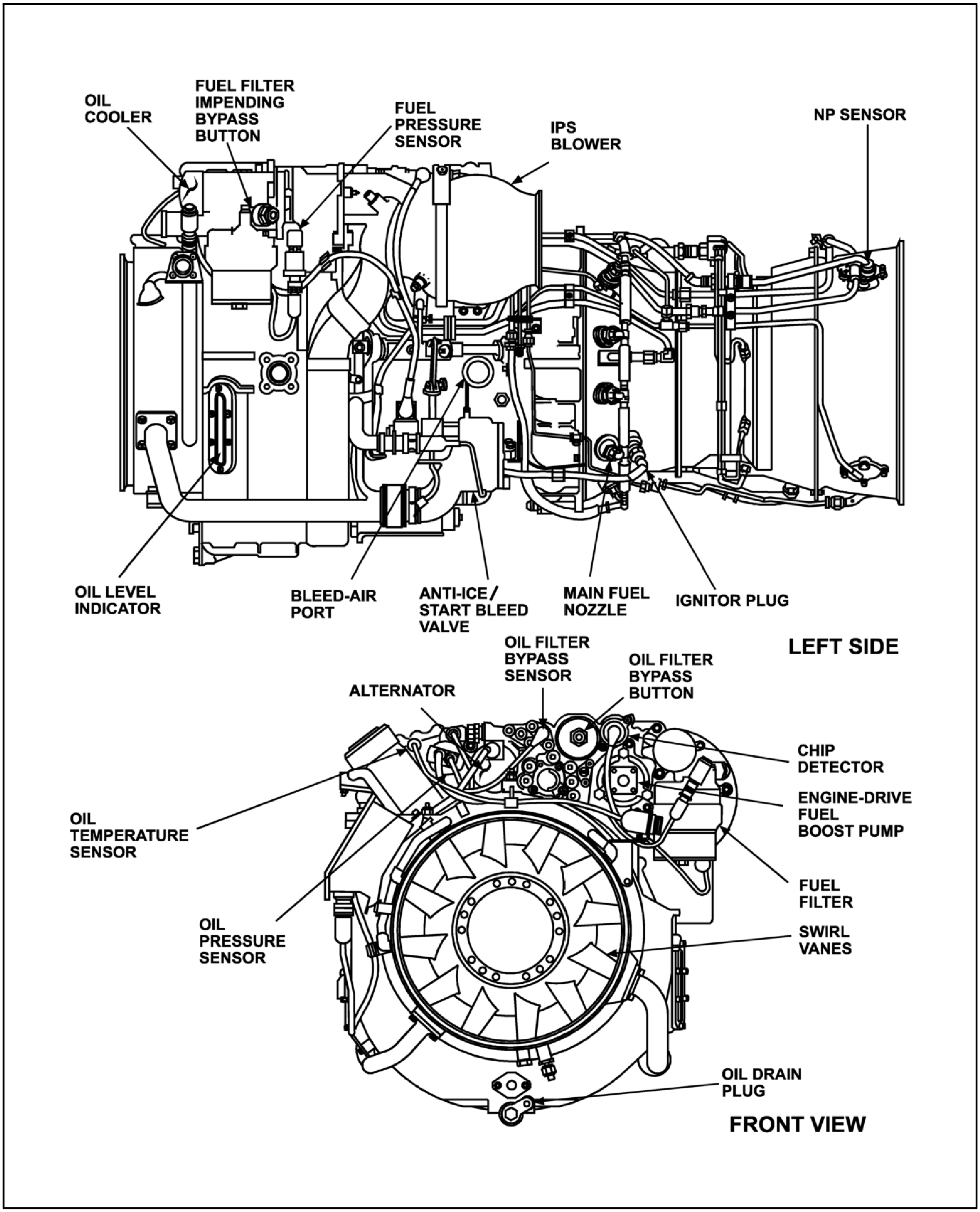


Figure 2-1. Engine, T700-GE-401C Profile (Sheet 1 of 3)

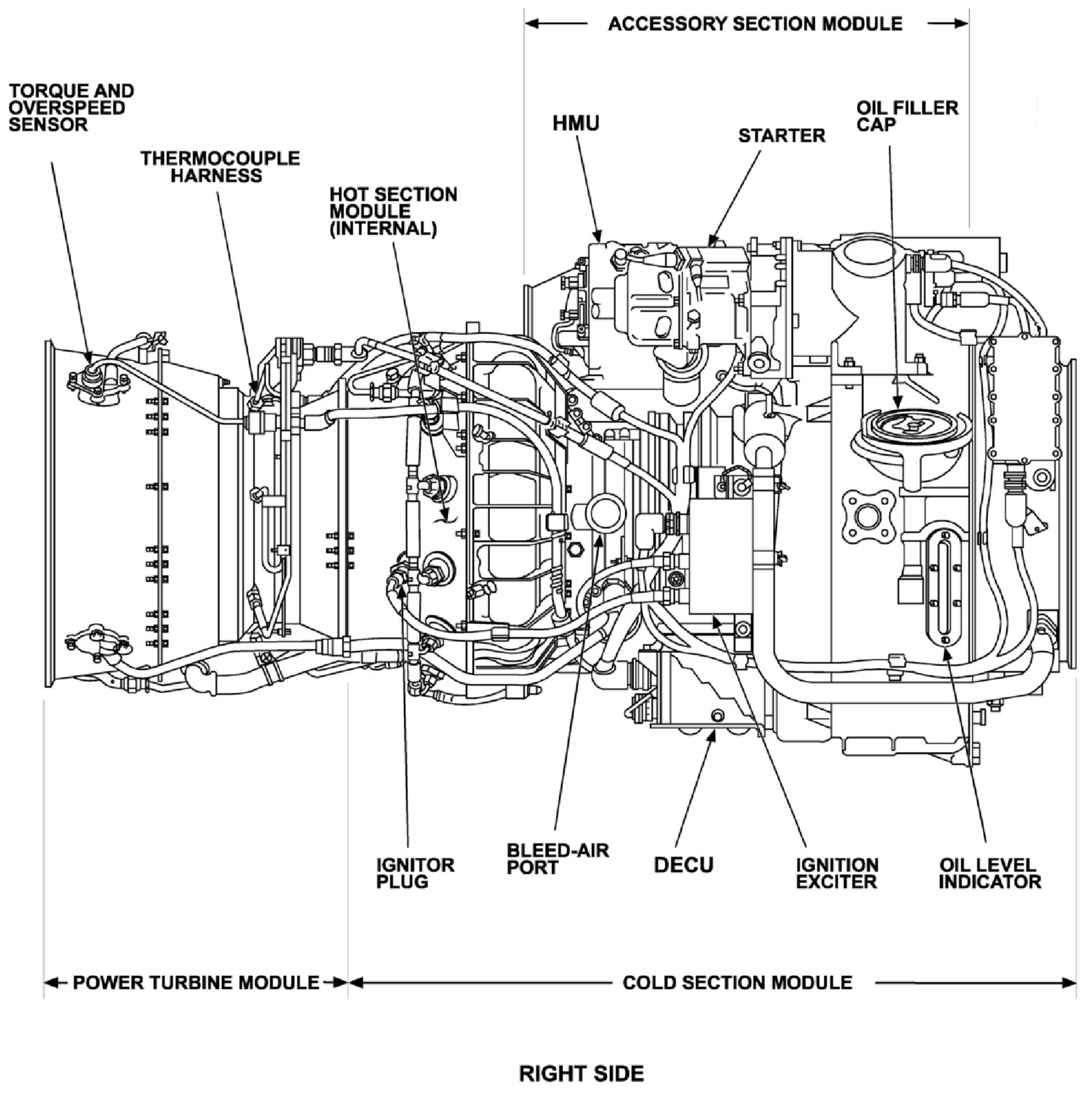


Figure 2-1. Engine, T700-GE-401C Profile (Sheet 2)

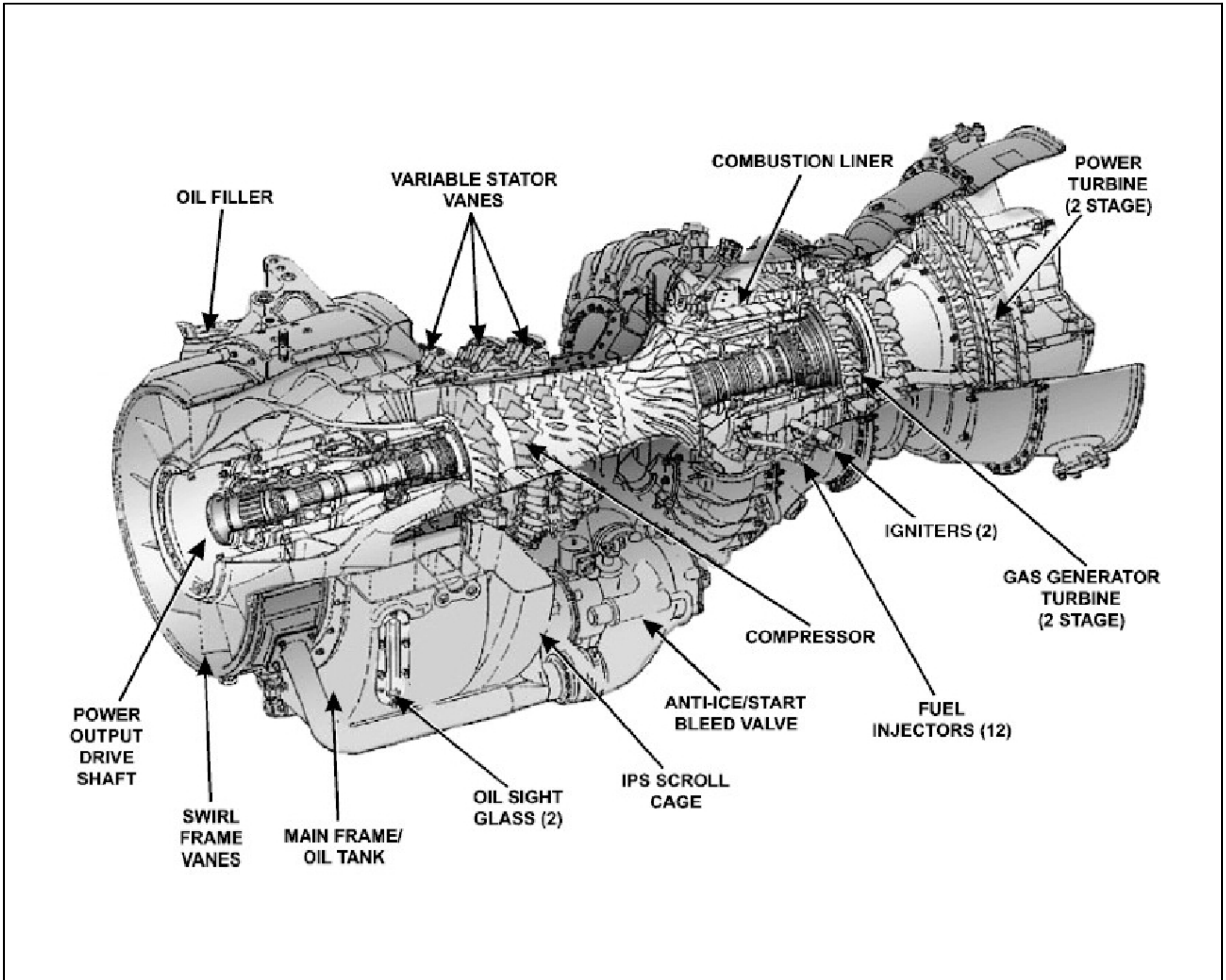


Figure 2-1. Engine, T700-GE-401C Profile (Sheet 3)

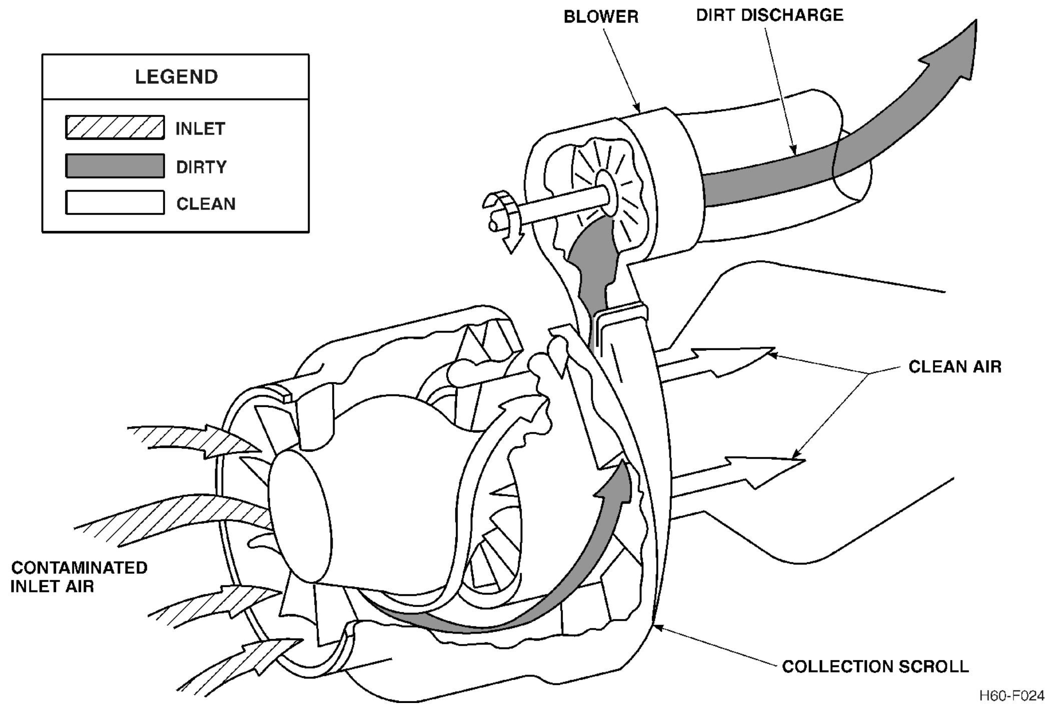


Figure 2-2. Inlet Particle Separator Airflow

2.1.1.7 Power Turbine

The N_p turbine has two stages that turn the power turbine drive shaft. The shaft is coaxial, turning inside the gas-generator turbine drive shaft. It extends through the front of the engine where it connects to the high speed shaft, which in turn connects to the input module. The power turbine is comprised of the power turbine rotors, power turbine drive shaft, power turbine case, and exhaust frame. Turbine Gas Temperature (TGT) is sensed between the gas-generator and power turbine.

2.1.1.8 Engine Airflow

Approximately 30 percent of the total airflow through the engine is used for the combustion process. The remainder is utilized for the following:

1. Compressor inlet temperature (T2) air.
2. Compressor discharge pressure (P3) air.
3. Combustor and turbine cooling.
4. Engine oil seal pressurization.

2.1.1.9 Main Frame and Accessory Section

The main frame contains the oil tank, oil level sight gauge, and accessory gearbox (AGB) supports. The accessory section mounts to the rear of the main frame at the 12 o'clock position, above the scroll case. The AGB is driven by a rotor via a radial drive shaft from the N_g turbine drive shaft. The rear face provides drive pads for the engine starter, HMU, IPS blower, and overspeed and drain valve (ODV). The front face provides drive pads for the alternator and engine-driven fuel boost pump. Mounting cavities are provided for the lube and scavenge pump and chip detector. Face-ported pads are supplied for the oil cooler, fuel filter, and oil filter. Cored passages in the AGB housing convey fuel and oil between components.

2.1.2 Engine Control System

The engine control system (Figure 2-3) includes all control units necessary for the proper and complete control of the engine to maintain a constant N_p/N_r. The major components are the hydromechanical control unit (HMU), overspeed and drain valve (ODV), digital electronic control unit (DECU), an engine–driven alternator, and a series of fuel flow control valves. Basic system operation is governed through the interaction of the DECU and HMU. In general, the HMU provides gas–generator control while the DECU trims the HMU to satisfy the requirements of the power turbine load and reduce pilot workload. The engine control system functions automatically, with no action required of the pilot after starting.

2.1.2.1 Engine Control Quadrant

The engine control quadrant (Figure 2-4) consists of two power control levers (PCL), two fuel selectors levers, two engine T-handles, and a rotor brake interlock. A starter button is located on each PCL. The PCL has four positions (OFF-IDLE-FLY-LOCKOUT).

With the PCL in the OFF position, the Power Available Spindle (PAS) mechanically shuts off fuel at the shutoff valve, within the HMU. Once the PCL is moved to the idle position, the HMU automatically controls start sequence fuel flow allowing the engine to achieve self–sustaining combustion. Placing the PCL in the FLY detent sets the maximum level of power that could be supplied, if demanded. If the PCL is momentarily advanced to LOCKOUT and then retarded, the PCL is used to manually control N_p and N_g. TGT limiting, N_p governing, and load sharing functions are deactivated and must be manually controlled. The N_p overspeed protection system is retained when in LOCKOUT via a direct link between the DECU and ODV. To return to automatic engine control, the PCL must be moved to IDLE, then returned to FLY.

A solenoid on the quadrant activates a mechanical locking device to prevent the PCLs from being advanced above IDLE with the rotor brake on. If the rotor brake is released, the solenoid energizes, and unlocks the PCLs. An override tab is provided on the quadrant, should the solenoid fail. This allows the PCLs to be advanced above ground idle by pulling down on the override tab.

2.1.2.2 Load Demand System

With the PCL in FLY, the HMU responds to collective position, through a load demand spindle (LDS) to automatically control engine speed and to provide required power. When the PCL is moved to LOCKOUT and then to some intermediate position, the engine will still vary power in response to collective position.

2.1.2.3 Engine Fuel System

The engine fuel system consists of the engine–driven fuel boost pump, fuel filter, HMU, and ODV.

2.1.2.4 Engine-Driven Fuel Boost Pump

The engine-driven fuel boost pump mounted on the forward side of the AGB is designed to:

1. Provide reliable suction feed from the aircraft fuel tank to the engine minimizing vulnerability and fire hazard in the event of damaged fuel lines.
2. Provide discharge pressure to satisfy the minimum inlet pressure requirement of the HMU or high–pressure fuel pump.

2.1.2.5 Engine Fuel Filter

The engine fuel filter provides filtration of solid particulate matter, but does not filter water. Fuel enters the filter inlet ports from the engine–driven fuel boost pump and is then routed to the HMU high–pressure fuel pump.

When a pressure differential across the fuel filter is sensed, the impending bypass pressure differential indicator (PDI) extends. The impending bypass PDI cannot be reset until the filter element and bowl are removed and the indicator is reset internally. The electrical bypass switch is activated by a pressure signal as the bypass valve opens. Once the filter is bypassed, the #1/#2 FUEL FLTR BYPASS caution will appear.

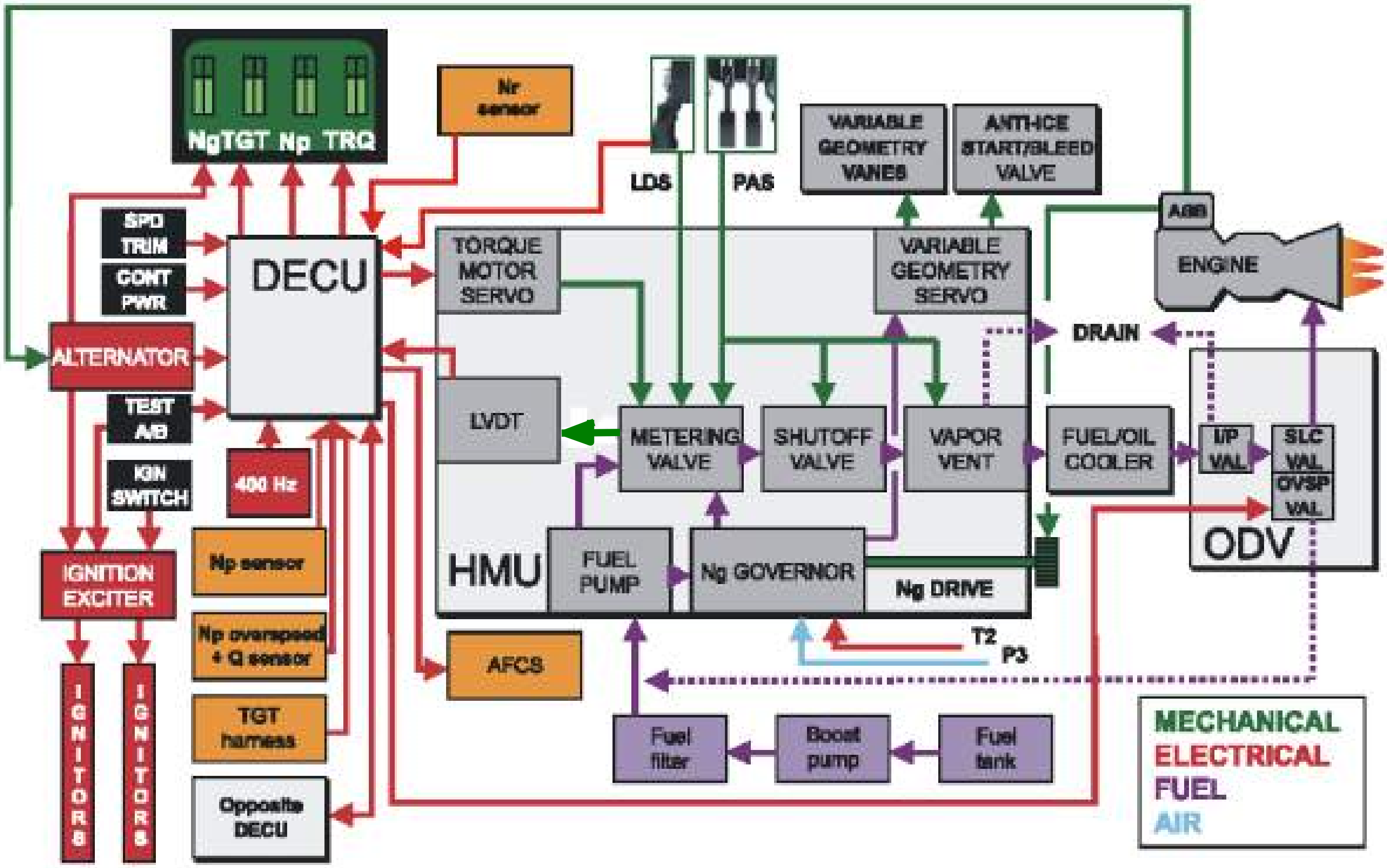


Figure 2-3. Engine Control Block Diagram

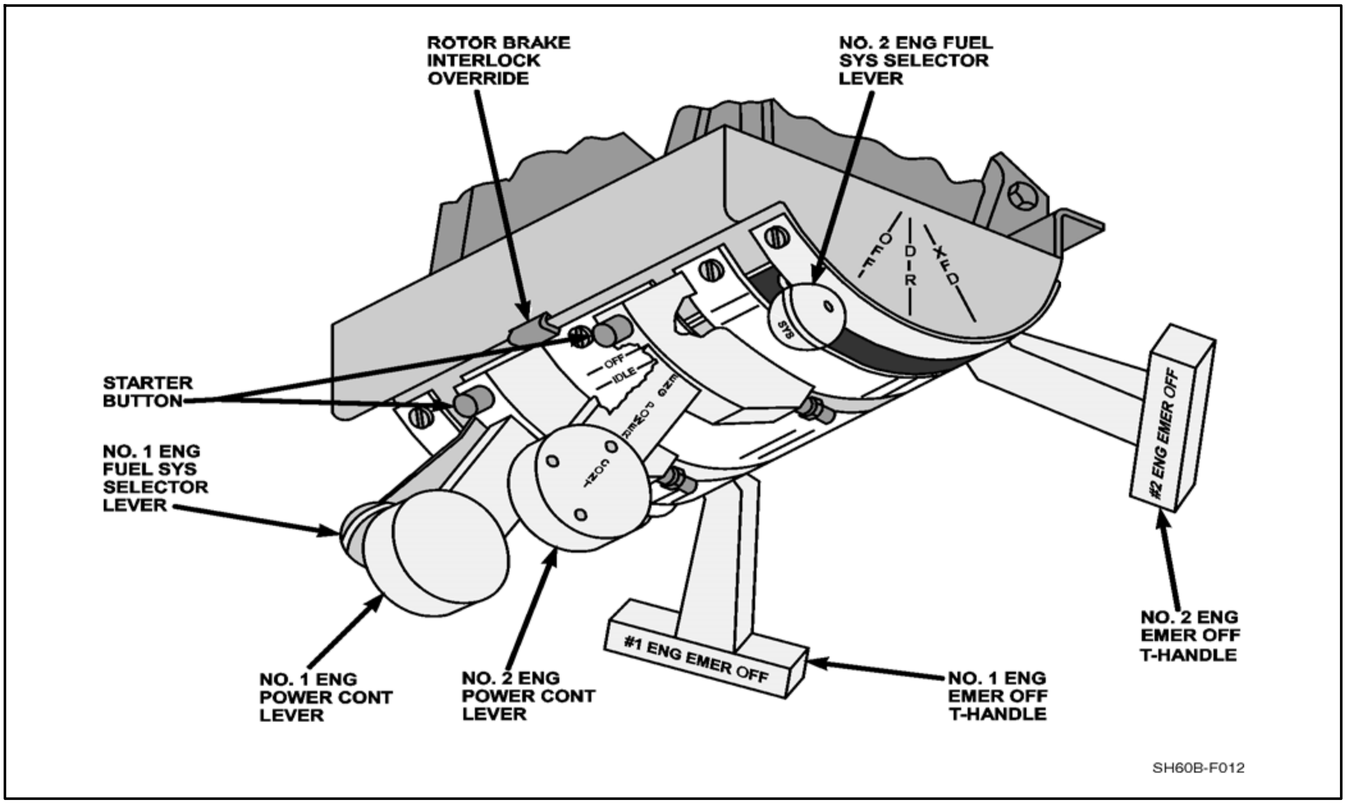


Figure 2-4. Engine Control Quadrant

2.1.2.6 Hydromechanical Control Unit

The Hydromechanical Control Unit (HMU), mounted on the aft center of the AGB, receives filtered fuel through a cored passage. It contains a high-pressure fuel pump, N_g governor, metering valve, linear variable displacement transducer (LVDT), torque motor servo, variable geometry vane servo, vapor vent, and shutoff valve.

The fuel enters the high-pressure engine-driven pump in the HMU, which provides high pressure fuel for efficient engine operation. The fuel leaves the pump and passes through the metering valve and shutoff valve, and is then directed through an external line to the oil-to-fuel heat exchanger. Some fuel is tapped off to operate various servos in the HMU for the following:

1. Positioning a metering valve to assure proper flow to the engine.
2. Positioning a servo piston that actuates the variable geometry vane servo and start bleed valve.
3. Amplifying various signals (T_2, P_3, N_g) which influence fuel flow and variable geometry servo position.

2.1.2.7 Inputs to the HMU

The HMU responds to two mechanical linkages from the cockpit and one electrical signal. The first mechanical input from the LDS directly coordinates N_g speeds to the approximate power required by the rotor system based on collective position. The second mechanical input is through the PCL. The position of the PCL manipulates the PAS at the HMU setting the desired power setting. A third input is in the form of an electrical signal from the DECU which actuates the torque motor servo in the HMU to precisely trim N_g speed for power turbine control and load sharing.

HMU receives cockpit inputs from the collective via the LDS and the PCL via the PAS. The HMU responds to the PCL for:

1. Fuel shutoff.
2. Setting engine start fuel flow with automatic acceleration to ground idle.

3. Setting permissible N_g up to maximum.
4. Fuel priming.
5. DECU override capability (LOCKOUT).

The HMU also responds to T2, P3, and N_g. These inputs aid the HMU in controlling variable stator vanes and anti-ice/start bleed valve position during engine start and normal operation, reducing the chance of compressor stall.

2.1.2.8 HMU Operation

The HMU operates as a conventional gas-generator power control when there is no input to the torque motor from the DECU. The HMU provides fuel scheduling for minimum flow, maximum flow, and variable stator vane control. Maximum and minimum metering valve stops provide absolute fuel flow limits.

The HMU fuel metering system controls fuel flow to the engine during all operating conditions. Fuel enters the high-pressure engine-driven fuel pump in the HMU, which provides adequate high pressure fuel for efficient engine operation. After fuel leaves the high pressure fuel pump, it is routed to the metering valve. The metering valve schedules engine fuel flow commensurate to current power demand and is trimmed to the required level by the torque motor servo in response to DECU signals. The HMU, via the LVDT, then provides a feedback signal to the DECU to null the torque motor servo input, stabilizing metering valve movement and preventing engine oscillation/hunting. Excess fuel is routed back to the pump inlet.

A nonadjustable topping setting controls maximum N_g during cold ambient operation and maximum TGT in the event of an electrical control system failure. If the N_g servo within the N_g governor reaches a position, corresponding to an overspeed, a spring-loaded ball valve ports fuel pressure causing the minimum pressure valve to secure flow to the engine. The N_g overspeed valve is set to trip at 110 percent ±2 percent N_g.

The PAS sets a maximum available N_g. Placing the PCL in FLY allows N_g to reach a setting that provides intermediate power. Collective movement adjusts available N_g to a power level approximately equal to the rotor load demand power. The actual level of engine power in FLY is normally more than required by the helicopter. This schedule is intentionally placed at a higher-than-required power level for two reasons:

1. Fail-safe to high power. The torque motor, when energized, is designed to reduce the schedule to the desired power level. Therefore, loss of torque motor electrical current causes the schedule to return to the highest power level. A schedule that is biased high due to engine electrical failure does not cause power limiting and can be manually retarded to a more desirable level using the PCL. With all engine protection functions in the HMU operational, neither engine damage nor stall can occur during or following loss of electrical signal to the torque motor.
2. Power available with one engine inoperative (OEI). In the event of a failure of one engine, the remaining engine's gas generator can increase power sufficiently up to its limit (contingency power) to carry the load at the given LDS setting. A load demand signal is introduced to the HMU through the LDS. When the LDS is reduced from its maximum setting by adjusting the collective, the N_g is reset from the PAS setting to provide immediate and accurate gas generator response. This new N_g setting is trimmed by the DECU to satisfy the N_p governing and load sharing functions.

The HMU provides:

1. Rapid engine transient response through collective compensation.
2. Automatic fuel scheduling for engine start.
3. N_g overspeed protection. The HMU mechanically limits N_g to 110 percent ±2. If the N_g servo, within the N_g governor, reaches a position corresponding to an overspeed a centrifugal valve secures fuel flow to the engine. Once the overspeed condition has passed, the valve re-opens, allowing normal operation to resume once the engine is primed and restarted.
4. N_g governing. The HMU receives T2, P3, and N_g inputs from their respective sensors, which are used to schedule fuel for minimum flow, maximum flow, and variable geometry vane control.

5. Acceleration limiting. The N_g governor ensures any PCL motion will result in safe engine operation and will not cause engine damage. Except for intentional shutoff of the PCL, an inadvertent shutdown will not occur during PCL motion.
6. Flameout and compressor stall protection. The HMU adjusts variable-geometry vane position and opens the anti-ice/start bleed valve to prevent compressor instability.

2.1.2.9 Overspeed and Drain Valve

When the PCL is advanced to IDLE during engine start, the shutoff valve in the HMU opens and allows metered fuel to flow to the Overspeed and Drain Valve (ODV) inlet. The ODV has four main functions:

1. Provides main fuel flow to the 12 fuel injectors during engine start and operation.
2. Purges the main fuel manifold overboard, after engine shutdown, through a shutoff and drain valve to prevent coking of the fuel injectors.
3. Traps fuel upstream, which keeps the fuel/oil heat exchanger full, so that system priming is not required prior to the next start.
4. Returns fuel back to the HMU if the N_p overspeed is energized or if the DECU hot start preventer is activated.

2.1.3 Engine Electrical System

2.1.3.1 Alternator

All essential engine electrical functions are powered by the alternator. The engine contains separate windings providing AC power to the Ignitor assembly, DECU, and N_g signal to the VIDS.

2.1.3.2 Digital Electronic Control Unit

The Digital Electronic Control Unit (DECU) resets the HMU within acceptable engine limits to maintain N_p governing while automatically limiting TGT.

The DECU is mounted below the compressor casing. The forward face of the DECU projects into the collection scroll case of the IPS where it is cooled by scavenge airflow. Four connectors provide for interconnection with the other engine control components, airframe systems, and the diagnostic equipment. The control parameters of the DECU are:

1. N_p sensing (governing).
2. N_p overspeed and torque sensing (load sharing, cockpit torque indication, and N_p overspeed protection).
3. TGT monitoring (temperature-limiting circuit).

The DECU receives the following inputs from the cockpit:

1. ENGINE SPD TRIM switch.
2. CONTGCY PWR switch.
3. ENG OVERSPEED TEST A and B buttons.

The DECU receives the following input signals from the helicopter:

1. Torque from the other DECU.
2. N_p demand.
3. 400-Hz backup power.
4. HMU (LVDT).

The DECU sends the following signals to the cockpit:

1. Torque.
2. N_p.
3. TGT.
4. Contingency power.

2.1.3.3 DECU Operation

During normal operations, the DECU performs the following functions:

1. N_p Governing – The N_p sensor located on the left side of the power turbine section provides an N_p signal to the DECU. Actual N_p is compared to a reference N_p to compute a speed error input signal for use in electrical control computation.
2. N_p Overspeed Protection – The N_p overspeed system is composed of redundant circuits, which rely on a signal from the N_p overspeed and torque sensor located on the right side of the power turbine section. The overspeed system is actuated at 120 percent N_p. When N_p exceeds 120 percent, a signal is sent from the DECU to the ODV, diverting fuel to the inlet of the HMU, causing engine flameout.

WARNING

A popped NO. 1 ENG OVSP or NO. 2 ENG OVSP circuit breaker shall not be reset in flight. Resetting a popped NO. 1 or NO. 2 ENG OVSP circuit breaker may initiate an engine overspeed signal and result in engine failure.

3. TGT Limiting – Measured TGT is compared to a fixed reference. When temperature is above the reference, a signal is generated to reduce fuel flow. When TGT approaches 851 °C, the DECU prevents any further increase in fuel flow to the engine. The intermediate range power (IRP) limiter will prevent this at 839 °C ±10 °C. If power demand is increased further, N_p/N_r will droop below 100 percent; N_p governing will be sacrificed to protect the engine against over temperature.
4. Engine Load Sharing – Torque signals are compared between the two engines via the respective DECUs. A torque error signal is generated if one engine torque is less than the other. The torque matching system operates by increasing power on the lower torque engine, while not directly affecting the higher torque engine.
5. Engine Speed Trim – An ENG SPD TRIM switch, located on the upper console, with positions INCR and DECR, controls the N_p of both engines simultaneously. There is no individual engine trim capability. The ENG SPD TRIM switch supplies a reference electrical signal to the DECUs for controlling N_p as required between 96 percent and 101 percent N_p.
6. Contingency Power – The TGT limit can be increased by placing the CONTGCY PWR switch on the collective to the ON position. This sends a signal to the DECU to allow TGT to increase to 903 °C however; the maximum contingency range power (CRP) limiter will prevent further increase in fuel flow to the engine at 891 °C ±10 °C. The #1/#2 ENG CONT PWR ON advisories indicate that contingency power has been selected. Placing the CONTGCY PWR switch to the ON position automatically deactivates the Environmental Control System (ECS).
7. N_p Overspeed Test – The test mode is activated by the ENG OVERSPEED TEST A and B buttons. When both switches are actuated, the N_p overspeed limit is re–referenced to 96 percent N_p. If power turbine speed decreases when either switch is pressed individually, the opposite test switch may be faulty.
8. DECU LOCKOUT operation capability – After being moved momentarily to LOCKOUT, the PCL is used to manually control N_g and N_p. As a result, engine power is no longer controlled by the DECU; it is set by PAS and LDS positions only. With the PCL in LOCKOUT the torque motor servo is disabled therefore deactivating TGT limiting, N_p governing, and load sharing. The N_p overspeed protection system is retained when in LOCKOUT. To regain automatic engine control, the PCL must be moved to IDLE then returned to FLY.

9. Cockpit Signals – Provides N_p, TGT, and torque signals to SDC for cockpit display.

10. Hot Start Prevention – Detects a hot start when TGT exceeds 900 °C with N_g below 60 percent and N_p below 50 percent and automatically stops fuel flow by tripping the ODV. Fuel flow is restored when TGT either decreases to 300 °C or after 25 seconds, whichever occurs first. Hot start prevention can be disabled by pressing and holding the ENG OVSP TEST A or B button for the duration of the start sequence. A self–test of the hot–start prevention system is performed while conducting a normal N_p overspeed system test.

11. Fault Diagnostic System – The DECU incorporates signal validation for selected input signals within the electrical control system. Signals are continuously validated when the engine is operating. If a failure has occurred, the failed component or related circuit will be identified by a pre–selected fault code. It is possible to have more than one fault code detected and each code should be treated as an individual fault. Fault codes will be displayed numerically on the engine torque indicator. Codes are displayed starting with the lowest code (4 seconds on and 2 seconds off), rotating through all codes, and then repeating the cycle. They can be suppressed/recalled by depressing either one of the ENG OVSP TEST buttons. Once the problem has been corrected, the codes will be cleared and may be verified after operating the engine at FLY. The fault codes displayed for approximately one minute when the following conditions are met:

    a. N_g less than 20 percent.

    b. N_p less than 35 percent.

    c. Other engine is shutdown.

    d. Aircraft 400 Hz power is available.

Note

If fault codes are not suppressed, DAFCS ground checks and blade fold will be inoperative.

12. 400–Hz airframe backup power capability – DECU functions receive 400–Hz AC power from the aircraft electrical system in the event of an alternator failure. A failure of either power supply by itself will have no impact on the DECU's ability to control the engine.

13. Transient Droop Improvement (TDI) – The TDI system is designed to initiate power turbine acceleration early by using anticipator signals from the TDI N_r sensor located on the left accessory module and a collective position sensor in the mixing unit. Circuits in the DECU increase fuel flow to the engine via the HMU torque motor servo at low torque settings when collective demand is increased rapidly or in the event of rapid N_r decay.

14. Auto Ignition System – When an N_p overspeed condition is reached and during the N_p Overspeed Test, the overspeed valve located in the ODV is opened to reroute fuel flow to the HMU inlet. When N_p drops below 120 percent, the auto ignition system closes the overspeed valve and turns on the igniters for 5 seconds to relight the engine. The N_p overspeed/auto–ignition system will continue cycling until N_p/N_r is controlled. A yaw kick may be experienced each time engine relights.

15. N_g Decay Rate Relight Feature – The auto–ignition system also includes an N_g–decay rate relight feature. If an engine flames out for any reason and exceeds a specified N_g deceleration rate, the auto–ignition system will turn on the igniters for five seconds in an attempt to relight the engine. The N_g–decay rate relight feature is disabled below 62 percent N_g.

2.1.3.4 Ignition System

The AC powered ignition system includes an ignition exciter unit, mounted on the right side of the engine main frame, and two igniter plugs. The ENGINE IGNITION switch, labeled OFF/NORM, is located on the upper console and serves both engines. When in the NORM position and either starter button depressed, the ignition system operates. Ignition is automatically shut off after the engine start motor is disengaged at starter dropout speed. In the OFF position, the system is de–energized, but engine motoring capability remains. Electrical power to the ignition exciter assembly is supplied by the engine-driven alternator during engine start or whenever the auto–ignition feature is activated.

2.1.4 Engine Operation Summary

2.1.4.1 Starting and Ground Idle

After engaging the starter, the shutoff valve in the HMU is opened by advancing the PCL to IDLE. Fuel is automatically scheduled by the HMU to a fixed flow at light-off and then an acceleration fuel flow as a function of N_g, P3, and T2 to idle. In idle, fuel flow is scheduled automatically. Power turbine speed is not governed with the PCL in IDLE since the engine will not produce enough power to drive the power turbine to 100 percent.

2.1.4.2 Takeoff and Climb

Before takeoff, the PCL is advanced from IDLE to the FLY detent. This allows the rotor head to accelerate to 100 percent N_p/N_r. N_g will increase as the PCL is advanced and will stabilize once the N_p governing speed of 100 percent is reached.

As collective pitch is increased, the LDS rotates within the HMU, demanding an increase in N_g to maintain 100 percent N_p/N_r. The DECUs adjust fuel flow to match torques and trim N_p/N_r to 100 percent. As N_g increases, the HMU schedule closes the anti-ice/start bleed valve, and the variable stator vanes open to increase airflow through the combustor and turbine. If collective pitch is increased significantly, TGT may approach the limiting value. When this occurs, the DECU prevents any further increase in fuel flow to the engine. If the power required is increased further, N_p/N_r will droop below 100 percent; N_p governing will be sacrificed to protect the engine against overtemperature.

2.1.4.3 Cruise and Descent

When collective pitch is reduced, the LDS will reduce fuel flow and N_g. The variable stator vanes will close slightly to optimize fuel consumption and preserve stall margin. Upon entering a descent, the same sequence of events reduces N_g to the point that the anti-ice/start bleed valve may begin to open. If the collective is fully lowered (e.g., autorotation power-off descent), both engine torque indications drop to zero by intervention of freewheeling units in the input modules. Once the engines are uncoupled from the rotor, N_r is free to accelerate above 100 percent. Both engines continue to govern N_p at 100 percent, ready to pick up the rotor load when collective is increased.

2.1.4.4 Summary

The engine control system functions automatically, with no action required of the pilot after starting. The system is functionally split between the HMU and DECU; the HMU provides functions essential to safe engine operation, while the DECU performs a fine trim to reduce pilot workload.

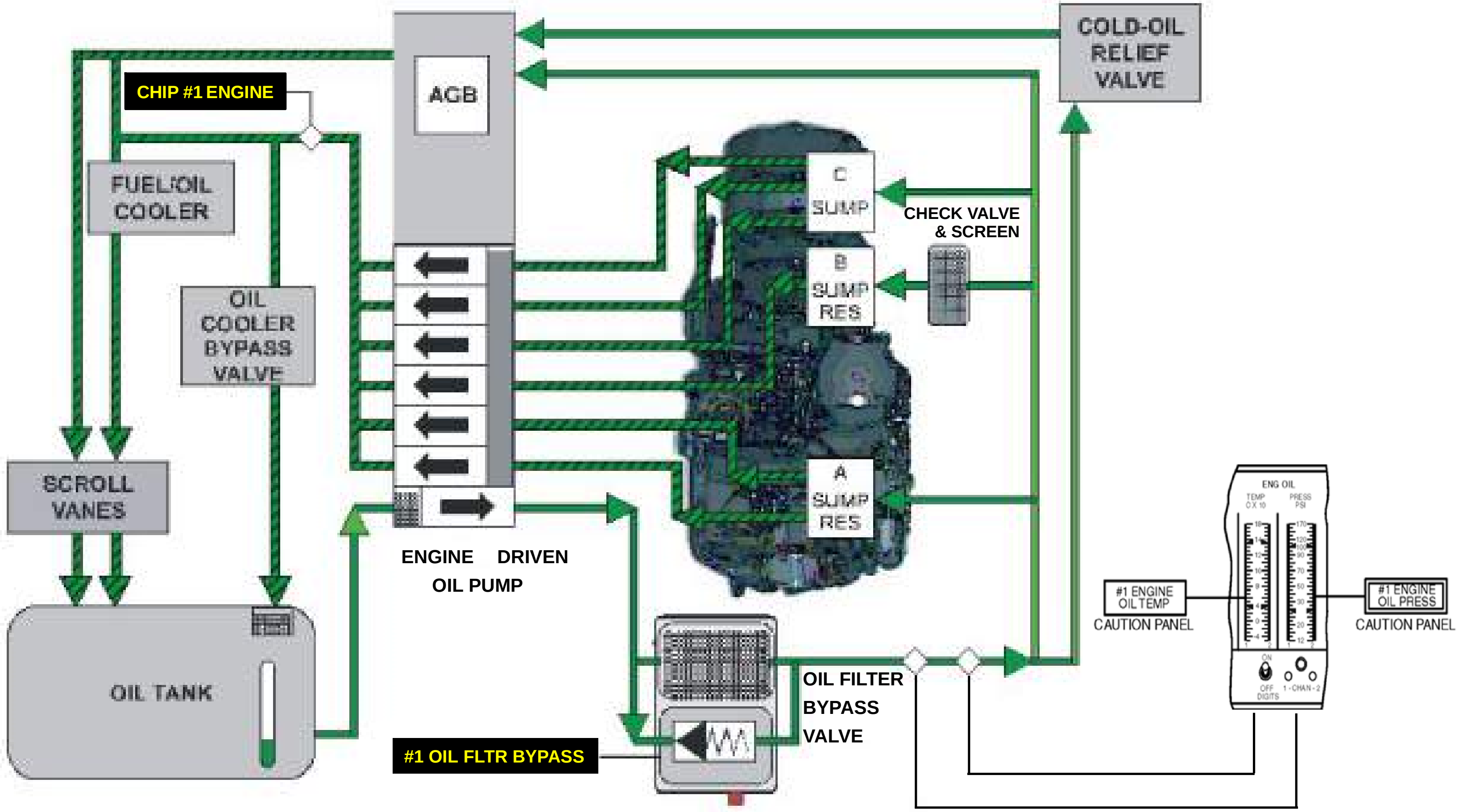
2.1.5 Engine Oil System

The engine oil system (Figure 2-5) is a self-contained, pressurized, recirculating, dry-sump system. It consists of the following systems and components:

1. Oil supply and scavenge system.
2. Oil filter and condition monitoring system.
3. Oil Tank.
4. Air/oil cooler (scroll vanes).
5. Fuel/oil cooler.
6. Chip detector.
7. Pressure and temperature indicators.

2.1.5.1 Oil Tank

The filler port is located on the right side of the engine (Figure 2-1). The oil level is indicated by a sight gauge on each side of the tank. Due to the design of the filler port, over-servicing is not possible. The scavenge pump returns oil from the sumps and AGB to the oil tank.



Note: The number of scavenge lines do not correlate to the number of bearings per sump.

Figure 2-5. Engine Oil System

2.1.5.2 Oil Supply and Scavenge System

Oil is picked up by suction created through the pressure element of the pump. It is then pressurized and flows through the oil filter into the passages in the AGB and the six main sump bearings. The engine is designed with two sets of oil jets to provide each main bearing with oil for lubrication and cooling. This redundancy provides for a brief period of operation following a malfunction or damage that totally interrupts the normal supply of oil. Scavenge oil flows through the pump inlet, electrical chip detector, fuel/oil cooler, the main frame, scroll vanes, and into the oil tank. If the oil pressure drops below limits, a caution, marked #1 or #2 ENGINE OIL PRESS will illuminate.

2.1.5.3 Oil Filter

Oil is discharged from the oil pump and routed to a disposable filter element. As the pressure differential across the filter increases, the impending bypass PDI will pop out, providing a visual indication that the filter element needs to be replaced. If the pressure differential continues to increase, the oil filter bypass sensor switch will activate the #1/#2 OIL FLTR BYPASS caution, indicating an oil filter bypass. When operating with a partially clogged filter, the high-pressure differential across the filter will cause the bypass valve to open and the caution to appear. The impending bypass PDI has a thermal lockout below 38 °C to prevent the indicator from popping during cold-weather starting.

2.1.5.4 Oil Cooler

Scavenge oil is cooled before it returns to the tank by a fuel/oil cooler mounted on the forward face of the AGB (Figure 2-5). As oil from the chip detector passes through the oil cooler, it is cooled by transferring heat from the oil to fuel.

After passing through the oil cooler, oil enters the top of the mainframe, where it flows through the scroll vanes that function as an air/oil cooler. This further cools the oil and heats the vanes for full-time anti-icing.

2.1.5.5 Engine Chip Detector

The chip detector consists of a housing with an integral magnet and electrical connector. The detector attracts ferrous metal particles at a primary chip-detecting gap. Chips are detected when this gap is bridged. A signal is then sent to the cockpit to illuminate the CHIP #1 ENGINE or CHIP #2 ENGINE caution.

2.1.6 Oil Temperature and Pressure Monitoring System

Engine oil system pressure and temperature sensors are mounted on the left forward face of the AGB. These sensors provide a signal to the VIDS for oil pressure and temperature indications. The #1/#2 ENGINE OIL PRESS and #1/#2 ENGINE OIL TEMP cautions are activated by the VIDS.

2.1.7 Engine Start System

The pneumatic start system (Figure 2-6) uses an air turbine engine start motor for engine starting. System components consist of an engine starter, start valve, check valves, controls, and ducting. One of two pneumatic sources may be selected as the source of air for engine starts: APU or engine crossbleed air.

The AIR SOURCE ECS/START switch is a three-position toggle switch that selects the source of air pressure for engine start and ECS operation. When the start button is pressed, air from the selected source is directed through the start valve to the engine starter. The #1/#2 ENGINE STARTER advisory will go on. The anti-ice/start bleed valve remains open to reduce backpressure and prevent compressor stall until N_g reaches approximately 90 to 94 percent (OAT-dependent). The #1/#2 ENG ANTI-ICE ON advisory will remain until the anti-ice/start bleed valve is energized closed.

As the engine alternator begins to turn, it supplies electrical power to the ignition exciter. Ignition will continue until starter dropout occurs (52 to 65 percent N_g). Once the starter drops out, the #1/#2 ENGINE STARTER advisory will disappear. If the starter fails to drop out automatically, it may be disengaged by pulling down on the PCL, pulling the circuit breaker or removing the air source. A malfunctioning starter may be overridden by manually holding in the starter button until N_g reaches 52 to 65 percent N_g.

2.1.7.1 Engine Start, APU

The APU provides bleed air and electrical power for engine starting. The APU will provide pneumatic power for engine start regardless of AIR SOURCE ECS/START switch position.

2.1.7.2 Engine Start, Crossbleed

Crossbleed engine starts are used when it is desired to start the other engine with the bleed air from the operating engine. The AIR SOURCE ECS/START switch must be placed to ENG and the operating engine must be at a minimum of 94 percent N_g or maximum N_g that can be safely attained. Pressing the starter button will simultaneously open the start valve on the engine not operating and the crossbleed valve on the operating engine.

2.1.8 Engine and Inlet Anti-Ice System

The engine and inlet anti–ice system (Figure 2-7). prevents ice buildup on the components of the engine inlet section. The system consists of the engine anti–ice start/bleed valve, mounted to the bottom of the compressor section, an inlet anti–ice valve and an inlet thermal switch, contained in the engine inlet cowling.

Hot air flow for anti–icing is distributed through two solenoid operated air valves. Both valves are held closed electrically and controlled by the ENGINE ANTI–ICE and DE–ICE MASTER switches on the upper console. When the engine inlet anti–ice valve is de–energized (valve open), bleed air is routed to a separate modulating valve in the engine inlet. When the DEICE MASTER switch is in the AUTO position, both solenoid valves are controlled by the ice detector.

There are three ways to anti–ice the engine:

1. Vent bleed air into the engine swirl vanes and engine inlet guide vanes (IGV) by the engine anti–ice/start bleed valve.
2. Vent bleed air into the airframe engine inlet by the engine inlet anti–ice valve.
3. Continuously pump engine oil through the scroll vanes.

2.1.8.1 Engine Anti–Ice/Start Bleed Valve

The engine anti–ice/start bleed valve provides 5th stage bleed air to the engine with anti–ice selected ON, and opens during engine starts. The valve remains open below approximately 90 percent N_g, to prevent compressor instability during starts. Above approximately 90 percent N_g, the anti–ice/start bleed valve closes, unless anti–ice is selected on, or the aircraft experiences a loss of electrical power. The temporary hang up of the engine variable geometry (VG) system at the anti–ice/start bleed valve may cause engine flameouts at low collective settings. The VG system is activated by fuel pressure from the HMU. To release the VG system quickly from any temporary hang–up condition while the collective is full down, the HMU will schedule maximum fuel flow to the VG actuator creating a diversion from the scheduled fuel flow to the engine. During these minimum fuel flow regimes, such as autorotations and quick stops, this diversion may be sufficient to flame out an engine.

A malfunctioning anti–ice/start bleed valve is indicated by any of the following:

1. Illumination of the ENG ANTI–ICE ON advisory light with above 90 percent N_g or above 94 percent N_g if OAT is 15° or greater.
2. No illumination of the ENG ANTI–ICE ON advisory light when N_g drops below approximately 88 percent N_g. (N_g may vary on a sliding scale depending on OAT).
3. No illumination of the ENG ANTI–ICE ON advisory light when the ENG ANTI–ICE switch is selected ON.
4. No rise in TGT when ENG ANTI–ICE switch is selected ON.

Note

With ENG ANTI–ICE ON, max torque available is reduced up to 18 percent per engine.

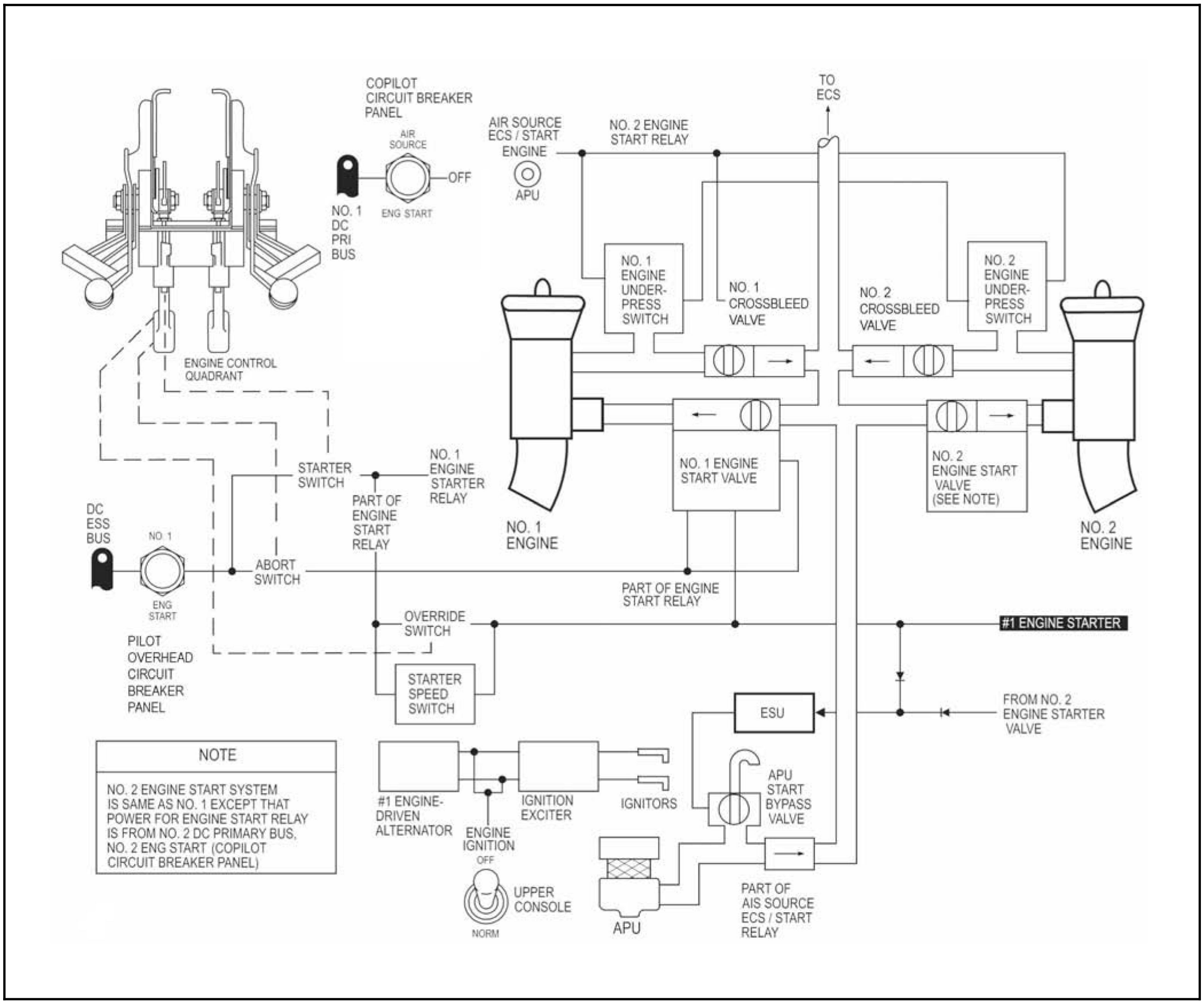


Figure 2-6. Engine Start Pneumatic/Electrical System, Block Diagram

2.1.8.2 Engine Inlet Anti-Ice Valve

The engine inlet anti-ice valve is a solenoid-actuated, modulating valve located in the engine inlet cowling. Bleed air is routed from the compressor, through the solenoid-actuated inlet anti-ice valve, then routed to the inlet cowling and inlet fairing.

The inlet thermal switch, mounted inside the engine inlet cowling, senses the air temperature inside the cowling. The NO. 1 or NO. 2 ENG INLET ANTI-ICE advisory will illuminate when bleed air heats the engine inlet to approximately 93 °C; however, full inlet anti-ice capability may not be available above 4 °C and will not be available above 13 °C. The inlet thermal switch does not have any input into or control over the inlet anti-ice valve or the Freon bellows.

With the NO. 1 and NO. 2 ENGINE ANTI-ICE switches OFF, the solenoid is energized and the valve is closed. With the NO. 1 and NO. 2 ENGINE ANTI-ICE switches ON, the engine inlet anti-ice valve is variably open based on OAT. With the NO. 1 and NO. 2 ENGINE ANTI-ICE switches OFF, the DE-ICE MASTER switch in AUTO, and ice is detected, the engine inlet anti-ice valve is variably open based on OAT.

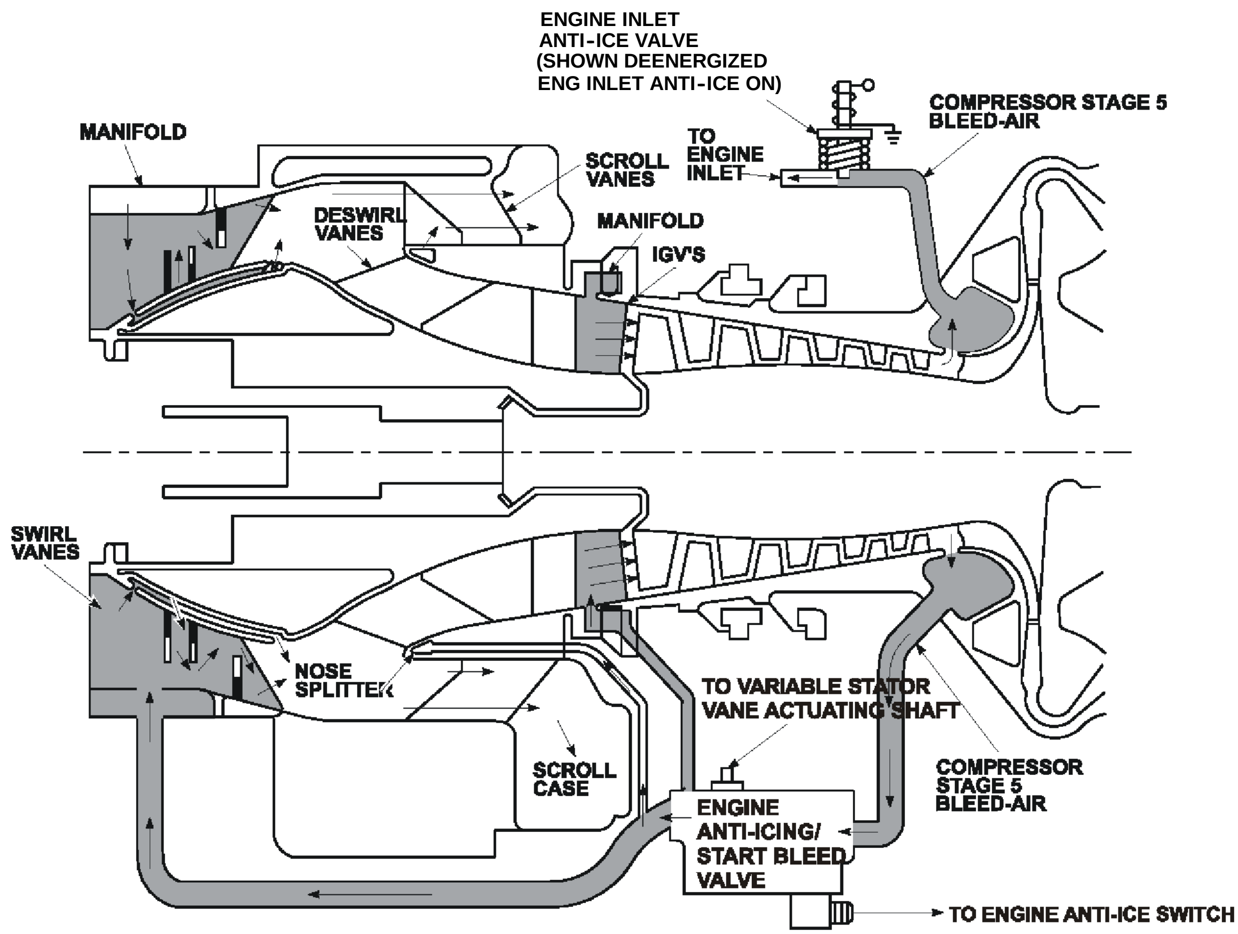


Figure 2-7. Engine and Inlet Anti-Ice System

Though the engine inlet anti–ice valve may be de–energized open, the release of fifth stage compressor bleed air into the engine inlet cowling and inlet fairing is ultimately controlled by the Freon–filled bellows, which reacts only to the ambient temperature. The bellows operates as follows:

1. Less than 4 °C, the valve is open and the ENG INLET ANTI–ICE ON advisories appear when inlet temperature reaches 93 °C.
2. Between 4 °C and 13 °C, the valve is controlled by a temperature compensating Freon–filled bellows. The bellows begin closing the valve when the OAT reaches 4 °C and should be completely closed by 13 °C.
3. Above 13 °C, the valve is closed and the ENG INLET ANTI–ICE ON advisories will extinguish when inlet cowling temperature drops below 93 °C.

WARNING

Illumination of the ENG INLET ANTI–ICE ON light when OAT is above 13 °C is an indication of a faulty engine inlet anti–ice modulating valve. The resultant loss of power could be a maximum of 49 percent when the engine anti–ice system is activated.

2.1.8.3 Engine Oil Circulation Through the Engine Scroll Vanes

An additional method of engine anti–ice protection is provided by the continual circulation of hot engine oil through the engine scroll vanes. The primary function of this circulation is to cool the hot engine oil providing anti–icing of the main frame. The air heated by the scroll vanes is vented overboard through the IPS and does not provide anti–ice protection to the air entering the compressor section of the engine.

2.1.9 Engine Parameter Sensing

2.1.9.1 N_p and Torque Sensing

Two N_p sensors are located on the top of the exhaust frame. The power turbine shaft is equipped with two pairs of teeth, which induce electrical pulses in the N_p sensors. These teeth permit measurement of the torsion or twist of the shaft, which is proportional to output torque, by producing a pulse of electrical current each time a shaft or reference tooth passes. The sensors are identical and interchangeable, but serve different functions. The left sensor provides an N_p signal to the DECU (used by the N_p governing circuitry) and the cockpit vertical instrument. The right sensor feeds the torque computation circuit and the N_p overspeed protection system. The electrical signal, which is conditioned in the DECU, provides a DC voltage proportional to the torque for cockpit indication and use by various engine subsystems.

2.1.9.2 N_g Sensing

The alternator supplies an N_g signal to the VIDS in the cockpit.

2.1.9.3 N_p Sensing

Two sensors are located in the exhaust frame. One sensor provides the N_p-governing and tachometer signal to the ECU. The other sensor feeds the torque computation circuit and the N_p overspeed protection system.

2.1.9.4 TGT Sensing

The thermocouple harness consists of seven thermocouples for measuring TGT. The thermocouples are joined in parallel and provide an average output that is provided to the DECU. The signal is relayed to the TGT VIDS from the DECU.

The T700–GE–401C engine TGT signal is biased — 71 °C when the engine alternator is operating (above approximately 28 percent N_g). The bias allows the higher rated T700–GE–401C TGT to be displayed on the VIDS.

2.1.10 Engine Instruments

2.1.10.1 Vertical Instrument Display System

The VIDS consists of a CDU and two Pilot Display Units (PDU) located on the instrument panel (Figure 1-8). The system furnishes all of the engine instrument readouts in the cockpit, including engine oil temperature and pressure, TGT, N_p, N_g, and torque. In addition, the system supplies instrument readouts for fuel quantity, transmission oil temperature and pressure, and N_r, which are discussed in applicable sections. These readings are shown by ascending and descending columns of multicolored lights (red, yellow, and green) measured against vertical scales. If the gauge contains red or yellow lights below the green lights, these lights will extinguish when the system indication reaches the lower green range segment. If the gauge contains yellow or red lights above the green range, the green as well as the yellow or red lights will stay illuminated when operating above the green range.

The CDU and PDUs contain photocells which automatically adjust the lighting of the indicators around a variable level set by the pilot with respect to ambient light level. If any of the three photocells should fail, the lights on the vertical scales of the PDUs and CDU will go out.

Note

The DIM knob on the CDU has a manual detent which will allow the pilot to set the lighting level to half intensity.

Two SDCs (Figure 2-8) take information from the NO. 1 and NO. 2 engines, N_r, transmission, and fuel quantity. The NO. 1 SDC receives:

1. NO. 1 engine sensor signals (oil pressure, oil temperature, turbine gas temperature, gas generator tachometer, torque, power turbine tachometer)
2. NO. 1 fuel quantity sensor signal
3. Main rotor speed sensor signal
4. NO. 2 engine power turbine tachometer signal
5. NO. 2 engine torque sensor signal.

The NO. 2 SDC receives:

1. NO. 2 engine sensor signals (oil pressure, oil temperature, turbine gas temperature, gas generator tachometer, torque, power turbine tachometer)
2. NO. 2 fuel quantity sensor signal
3. Main rotor speed sensor signal
4. NO. 1 engine power turbine tachometer signal
5. NO. 1 engine torque sensor signal
6. Main transmission oil temperature sensor signal
7. Main transmission oil pressure sensor signal.

Within each SDC, the associated sensor signals, except for NO. 1 and NO. 2 fuel quantity, main transmission oil temperature, and main transmission oil pressure, are conditioned to a common digital format for multiplexing. The fuel quantity and main transmission sensor signals are conditioned and multiplexed within the CDU. After the sensor signals have been conditioned and multiplexed, the sensor data is routed to latching circuits in the CDU and PDU. The latching circuits retain the last signal data until it is time to update. During update (twice per second), the latches activate lamp drivers that energize miniature lamps on the edge of the display modules. Light from the lamps is carried to the display panel face by fiber optic strips, giving visual analog and digital displays corresponding to the level of the sensed parameter. If either SDC fails, the applicable CHAN light on the CDU will illuminate, the pilot or ATO PDU will fail, and the corresponding instruments on the CDU will fail.

Both SDCs receive N_p and torque information from both engines as well as N_r; therefore, if the NO. 1 SDC fails, the pilot PDU will have N_r, N_p, and torque for both engines. The SDC receives DC power from the NO. 1 and NO. 2 DC primary buses through circuit breakers marked NO. 1 and NO. 2 DC INST on the ATO circuit breaker panel and AC power from the NO. 1 and NO. 2 AC primary buses through circuit breakers marked NO. 1 and NO. 2 AC INST on the center and the corner circuit breaker panels, respectively.

The following are the controls for the CDU and PDUs (located on the CDU):

1. LAMP TEST. When pressed, all lights on the CDU and the overspeed lights on the PDUs will illuminate, the digital readouts will display 888, and the 1 CHAN 2 failure lights will illuminate. When released, all lights and digits return to original readings.
2. DIM. This knob is used to control the intensity of the vertical scales and digits on the CDU and PDUs. It contains an override switch at the extreme clockwise end of the control range, allowing the pilot to manually set the CDU and PDUs to half intensity if the auto dim system fails.
3. DIGITS. The DIGITS Control switch, marked ON and OFF, is used to turn on or off the CDU and PDU digital readouts.
4. 1 CHAN 2. If a failure is detected in either SDC, the corresponding SDC CHAN 1 or 2 fail light will light.

2.1.10.1.1 Central Display Unit

The CDU contains 12 analog displays, 5 digital displays, and 2 failure lights. The CDU receives signal and power inputs from both the NO. 1 and the NO. 2 SDC. The multiplexed data signals applied from the SDC to the CDU contain engine oil temperature, engine oil pressure, TGT, and N_g information and are displayed on the CDU gauge. These parameters are displayed on the CDU analog scales. The TGT and N_g information is also displayed on CDU digital readouts. NO. 1 and NO. 2 fuel quantity signals from the NO. 1 and NO. 2 SDC, respectively, and main transmission oil temperature and pressure signals from the NO. 2 SDC are conditioned and multiplexed by the CDU for display on analog scales. In addition, total fuel quantity information is displayed on a digital readout. The fuel quantity and transmission temperature and pressure gauges are discussed in applicable sections.

1. Engine Oil System Gauges. Engine oil temperature and pressure gauges are provided for each engine. The oil temperature gauge scales are nonlinear and read from –50 to 180 °C. Index marks define the normal operating range. The oil pressure gauge scales are nonlinear and read from 12 to 170 psi. Index marks define the normal operating range.
   a. Engine Oil Temperature Gauge. The CDU displays oil temperature for both engines under the heading ENG OIL TEMP. The engine oil temperature gauge receives signals from the oil temperature sensor mounted on the front of the AGB.
   b. Engine Oil Pressure Gauge. The CDU displays oil pressure for both engines under the heading ENG OIL PRESS. The engine oil pressure gauge receives signals from the oil pressure transmitter mounted on the front of the AGB.

Note

The engine oil caution lights are triggered by the VIDS oil gauge positions. Therefore, caution light and gauge indication cannot be used as secondary indications to each other.

2. Turbine Gas Temperature Gauge. The CDU displays turbine gas temperature for both engines under the heading TGT. The two TGT nonlinear gauges indicate the turbine gas temperature of each engine and read from 0 to 950°C. There is an index at 925 °C. At the bottom of each TGT gauge is a digital readout for each engine. The TGT gauge receives signals from a thermocouple assembly located at the power–turbine inlet section of the engine.
3. Gas Generator, Turbine–Speed Gauge. The CDU displays gas generator, turbine–speed for both engines under the heading N_g SPEED. The gauges are nonlinear and read from 0 to 110 percent. Index marks define the normal operating range. At the bottom of each N_g gauge, there is a digital readout for each engine. The gas generator, turbine–speed gauge receives signals from the alternator mounted on the front of the AGB.

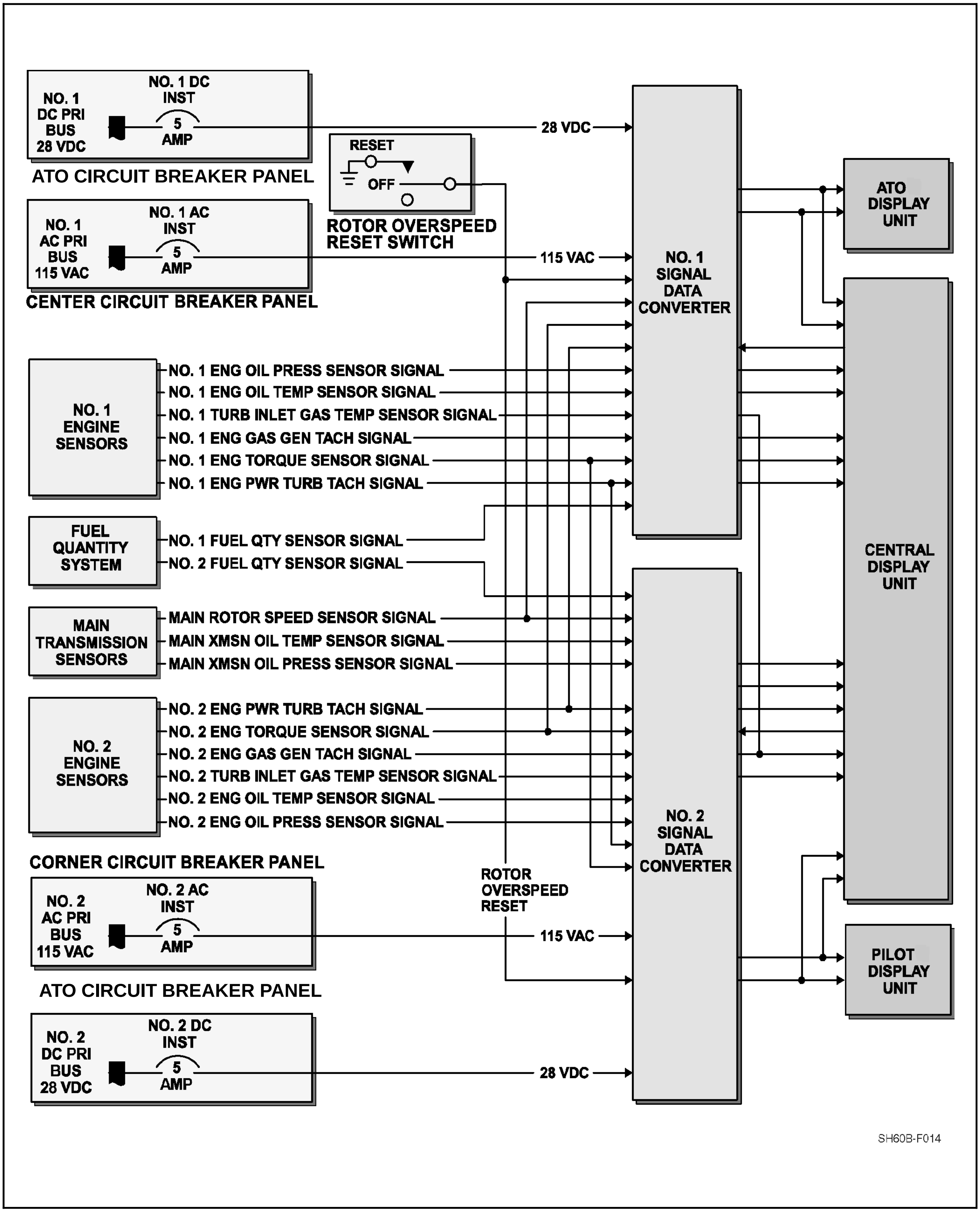


Figure 2-8. Signal Data Converter

Note

The #1 and #2 ENG OUT warning lights, located on the master warning panels, are tripped by the N_g VIDS indication automatically when N_g indication decreases below 55 percent.

The CDU failure lights, CHAN 1 and CHAN 2, are part of the VIDS fault detection circuit. A failure of any SDC or CDU processing circuit, CDU or PDU display driver module, or SDC logic power supply will cause the associated display channel to turn off or switch to the remaining SDC and will light the associated CHAN failure light. Failure of a lamp power supply within an SDC will cause every second display light on the CDU as well as all display lights on the corresponding PDU to go off.

Power to operate the CDU is provided from the NO. 1 and NO. 2 DC primary buses through circuit breakers, marked NO. 1 and NO. 2 DC INST; NO. 1 and NO. 2 AC primary buses through the SDCs.

2.1.10.1.2 Pilot Display Units

Two PDUs, each identical and interchangeable, contain indicators that display engine power turbine speed (N_p), rotor speed (N_r), and torque readings (TRQ) for each engine (Figure 1-8).

Each unit contains five analog displays, two digital displays, three RTR OVERSPEED indicator lights, a TEST switch, and a photocell. Instrument readings are shown by ascending and descending columns of multicolored lights with tracking arrows along the analog scales from 96 to 112 percent on the percent rpm indicator and 0 to 10 percent on the percent TRQ indicator. The tracking arrows for each display go on one at a time to coincide with analog scale indication. As one tracking arrow lights, the other goes off. The percent rpm display contains bottom segment turnoff, which turns off lower red and yellow lights. When the lowest green segment is reached, all red and yellow segments below the normal range will go off. Three RTR OVERSPEED lights go on from left to right when rotor speed is over 127 percent, 137 percent, and 142 percent, respectively. The lights are latched on and remain on even if rotor speed falls below the specified overspeed limits. The latch mechanism, located in the nosebay, is not affected by power loss or power interruption. The TEST switch on each display unit is used to check all vertical scale lamps and digital readouts on the associated unit. The photocells on the PDUs are used for automatic light level adjustment. The PDUs are powered by the NO. 1 and NO. 2 SDCs.

2.2 ROTOR SYSTEMS

The aircraft is configured with a single four-bladed main rotor and a 20° canted, four-bladed tractor tail rotor. The fully articulated main rotor head incorporates elastomeric bearings. The tail rotor is a hingeless crossbeam rotor of composite construction. An automatic electrically actuated main rotor blade fold is incorporated.

2.2.1 Main Rotor System

The fully articulated main rotor system consists of four subsystems: main rotor blades, hub, flight controls, and the bifilar vibration absorber. The four main rotor blades attach to hinged spindles and are retained by elastomeric bearings contained in a one-piece titanium hub. The elastomeric bearings are laminated rubber and stainless steel and enable the blades to flap, lead, and lag, and also permit the blade to move about its axis for pitch changes. Two bearings are used per blade.

The main rotor vibration absorber is mounted on top of the hub and consists of a four-arm plate with attached weights. Main rotor dampers are installed between each of the main rotor spindle modules and the hub to restrain lead and lag motions of the main rotor blades during rotation and to absorb rotor head starting loads. Each damper is supplied with pressurized hydraulic fluid from a reservoir mounted inside the main rotor shaft. The reservoir has indicators to monitor the fluid level and nitrogen precharge pressure.

Rotor control is provided by flight control hydraulic servos tilting the swashplate assembly, which moves control rods attached to each spindle. When the rotor is not turning, the blades and spindles rest on hub–mounted droop stops. Upper restraints called antiflapping stops limit flapping motion at low rotor rpm. Both stops engage as the rotor slows down during engine shutdown. When the main rotor is rotating above 35 percent, centrifugal force pulls the antiflapping assemblies outward and holds them in that position to permit flapping and coning of the blades. When the main rotor head is rotating between 55 percent and 60 percent N_r, centrifugal force pulls the droop stops out and permits increased vertical movement of the blade.

2.2.1.1 Main Rotor Head

The main rotor head (Figure 2-9) transmits the movements of the flight controls to the four main rotor blades. The main rotor head is supported by the main rotor shaft extension. The lower pressure plate, in conjunction with the main shaft nut, secures the shaft extension to the main shaft. The lower pressure plate also provides attachment for the scissors.

1. Swashplate. The swashplate has stationary and rotating discs separated by a bearing. It transmits flight control movement to the main rotor head through the four pitch control rods. The swashplate is permitted to slide on the main rotor shaft around the Teflon–coated uniball and tilt in any direction following the motion of the flight controls.

2. Pitch Control Rods. Four pitch control rods extend from the rotating swashplate to the blade pitch horn on each spindle. The pitch control rods transmit all movement of the flight controls from the swashplate to the main rotor blades. Each rod is ground adjustable for blade tracking.

3. Bifilar Vibration Absorber. This unit absorbs vibrations and stresses. The bifilar vibration absorber is a cross–shaped aluminum forging. A tungsten weight pivots on two points at the end of each arm. The bifilar is bolted to the main rotor hub.

2.2.1.2 Main Rotor Blade

Four main rotor blades are installed on the main rotor head. Each blade has a pressurized titanium spar, honeycomb core, fiberglass graphite skin, nickel and titanium abrasion strips, electrothermal deicing mats, and a removable swept–back blade tip cap. The 20° swept tips provide both sound attenuation and increased rotor blade efficiency. An electrothermal blanket is bonded into the leading edge for de–ice protection. A pressure indicator and servicing valve are installed at the inboard end of the blade. A titanium cuff provides the attachment of the blade to the rotor head. The spar of the main rotor blade is pressurized with nitrogen. If the blade is damaged, impairing the structural integrity of the spar or if a seal should leak, nitrogen will escape. The pressure will drop below the minimum and cause the Blade Inspection Method (BIM®) pressure indicator (Figure 2-10) to show a black or unsafe indication. The nickel and titanium abrasion strips bonded to the leading edge of the spar extend the useful life of the blades. Each blade is statically and dynamically balanced. This permits replacement of individual blades. Balance strips painted around the blade locate the hoisting points.

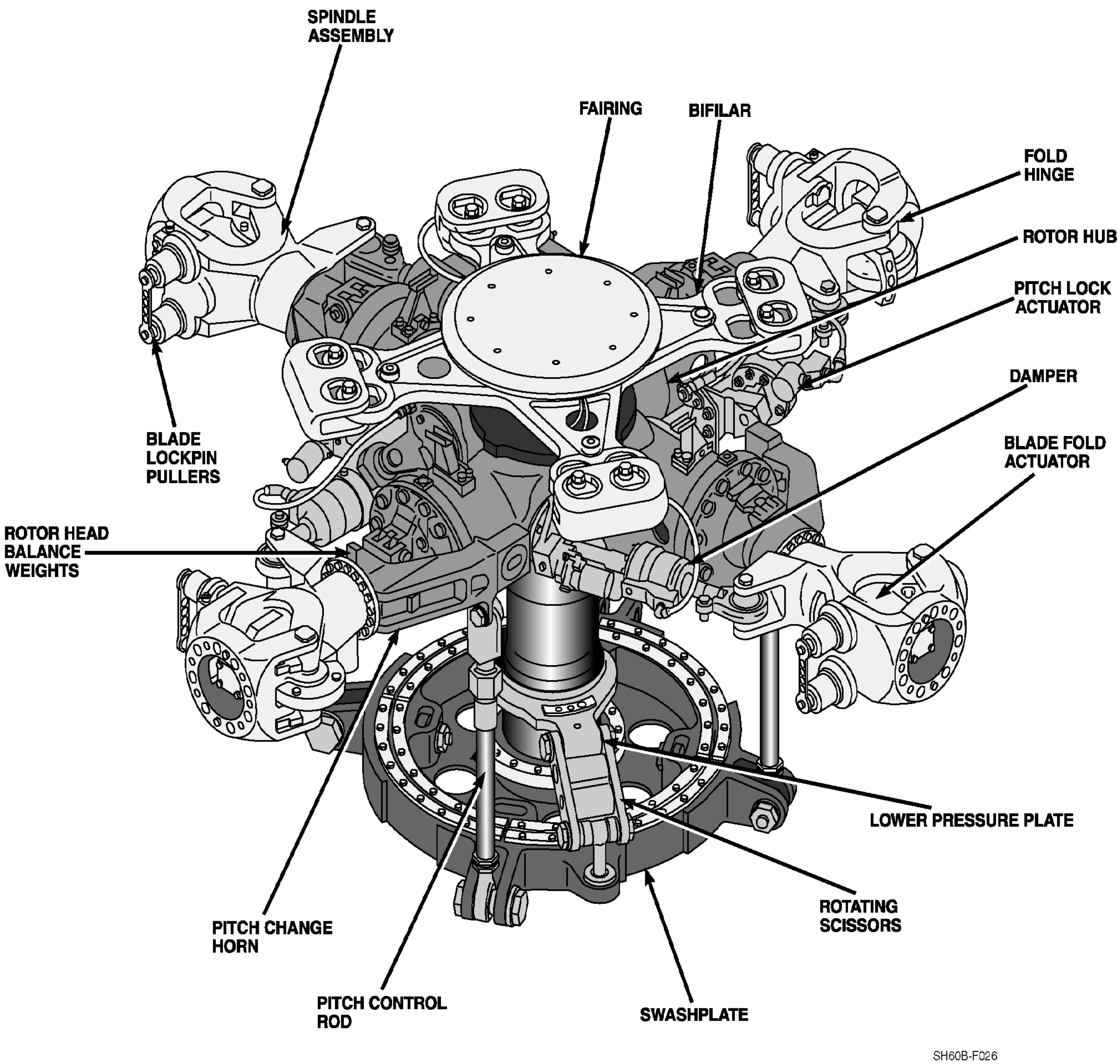


Figure 2-9. Main Rotor Hub Assembly

2.2.1.2.1 Pressure Indicator

The BIM® indicator (Figure 2-10) is installed in the back wall of the spar at the root of the blade. A color change indicates an unserviceable blade. The indicator compares a reference pressure built into the indicator with the pressure in the blade spar. When the pressure in the blade spar is within the required service limits, three white stripes are visible. If the pressure in the blade spar drops below the minimum permissible service pressure, the indicator will show three black stripes. A manual test lever is installed on each BIM® indicator to provide a maintenance check.

WARNING

If black is visible on the indicator, it may be an indication of blade damage that is a flight hazard. The cause of the black indication shall be determined prior to flight.

2.2.1.2.2 Blade Retention

Each rotor blade is connected to the rotor head spindle outboard of the blade–fold hinge and is attached by means of a bolted flange. The blade retention assembly provides means of attachment to the rotor hub, which allows interchangeability of the rotor blades. The retention assembly does not have to be removed to service or maintain the main rotor hub.

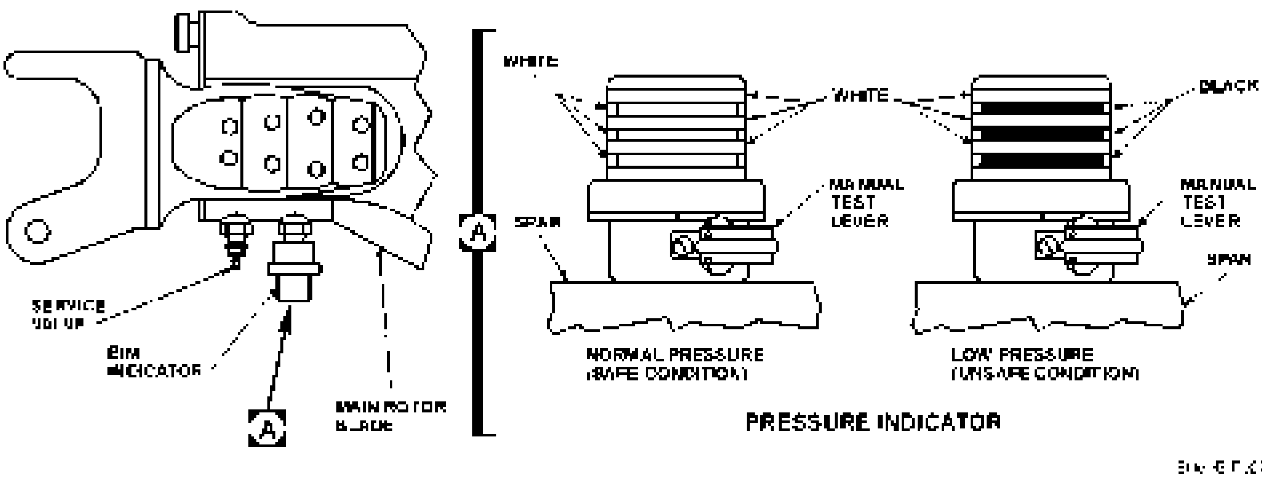


Figure 2-10. BIM® Indicator

2.2.1.2.3 Main Rotor rpm-Indicating System

The N_r gauge is located on each PDU on the instrument panel and indicates the speed at which the main rotor is turning. Three red warning lights on top of each PDU indicate varying degrees of rotor overspeed. The left light illuminates at 127 percent N_r, the middle at 137 percent N_r, and the right light at 142 percent N_r. Once the lights illuminate, they will remain on and must be manually reset on deck. Main rotor rpm is sensed in the right-hand accessory module of the main transmission and transmits a signal to each PDU which indicates speed in percent N_r. The N_r gauge is powered by the NO. 1 and NO. 2 DC primary buses through circuit breakers, marked NO. 1 and NO. 2 DC INST. Both circuit breakers are located on the ATO circuit breaker panel.

2.2.2 Tail Rotor System

A bearingless crossbeam tail rotor blade system provides antitorque action and directional control. The blades are of graphite and fiberglass construction. Blade flap and pitch-change motion are provided by deflection of the flexible graphite fiber spar, eliminating all bearings and lubrication. The spar is a continuous member running from the tip of one blade to the tip of the opposite blade. Electrothermal blankets are bonded into the blade-leading edge for deicing. The tail rotor head and blades are installed on the right side of the tail pylon, canted 20° upward. In addition to providing directional control and antitorque reaction, the tail rotor provides 2.5 percent of the total lifting force in a hover. With a complete tail rotor control failure, a centering spring in the tail rotor control system will provide a preset spring-loaded position for the tail rotor, equivalent to the antitorque requirements for a midposition collective power setting.

2.2.2.1 Tail Rotor Quadrant

The tail rotor quadrant (Figure 2-11), mounted on the tail gearbox, transmits tail rotor cable movements into the tail rotor servo. Two spring cylinders are connected to the quadrant. In the event a cable is broken, the spring tension allows the quadrant to operate normally. If a failure of a cable should occur, the quadrant then controls the opposite direction against spring tension and the related microswitch mounted on the quadrant will light the TAIL ROTOR QUADRANT light on the caution panel. The caution system for the tail rotor quadrant is powered by the NO. 1 DC primary bus through a circuit breaker marked TAIL ROTOR SERVO WARN and located on the ATO circuit breaker panel.

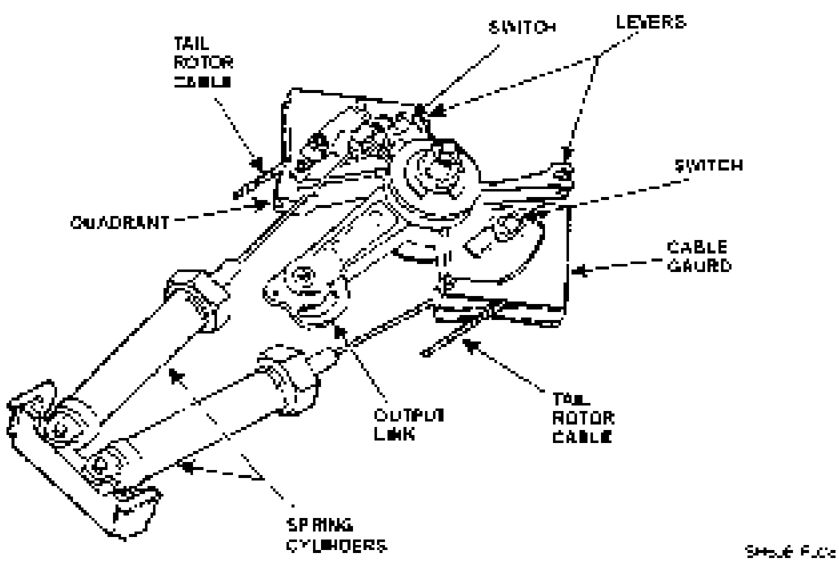


Figure 2-11. Tail Rotor Quadrant

2.2.2.2 Tail Rotor Head

The head consists of two titanium plates. The inboard plate forms a hub that is attached to the gearbox output shaft and retained by a shaft nut. The hub plates absorb axial thrust loads and bending moments and transmit torque to the rotor blades. The blade spars are clamped directly between the plates by retaining bolts. The pitch-control crossbeams are attached to a pitch-change actuating shaft, extending out from the center of the tail gearbox.

2.2.2.3 Tail Rotor Blades

The blades are built around two graphite composite spars, running from tip-to-tip and crossing each other at the center to form the four blades. The two spars are interchangeable and may be replaced individually. The blade spars are covered with cross-ply fiberglass to form the airfoil shape. Polyurethane and nickel abrasion strips are bonded to the leading edge of the blades. Blade-pitch changes are made by twisting the spar.

2.2.3 Rotor Brake System

The rotor brake system (Figure 2-12) is designed to hold the rotor during engine starting and with both engines at IDLE and to provide rotor shutdown. The system consists of a reservoir, master cylinder, gauge, relief valve assembly, accumulator, pressure switch, rotor brake advisory light, brake assembly, and disc. When the rotor brake is applied, the rotor brake interlock in the engine control quadrant prevents the ENG POWER CONT levers from being moved forward of the IDLE detent with the rotor brake on.

When the rotor brake lever is moved toward the apply position, pressure is built up in the lines and applied to the brake assembly. At the same time, pressure is applied to the rotor brake accumulator and a pressure switch (minimum 6 psi) to turn on the ROTOR BRAKE advisory light and set the ground IDLE quadrant lock. Back pressure in the accumulator is held by a spring as long as the rotor brake lever is in the applied position. For limited internal pressure leaks, the accumulator spring pressure will maintain the applied pressure to the brake pucks until the brake is released by returning the rotor brake lever to the off position, venting pressure back to the master cylinder reservoir. The rotor brake should not be applied with engine(s) operating and rotor head turning. The brake disc is mounted on the tail drive shaft output of the main gearbox. Teeth on the disc are utilized in the positioning cycle of the blade-fold system. The rotor brake advisory light system is powered by the DC essential bus through a circuit breaker marked ROTOR BRAKE on the overhead console circuit breaker panel.

2.2.3.1 Rotor Brake Master Cylinder

The rotor brake master cylinder, on the right side of the overhead console (Figure 2-12), provides pressure to the rotor brake assembly. With the master cylinder in the detent position (rotor brake lever off), the system is vented back to the reservoir. The hand pump reservoir serves as the rotor brake reservoir. A T-shaped rotor brake lever lock pin is provided to prevent inadvertent release once the brake has been applied. To set the pin, after the rotor brake lever is forward, rotate the pin 90° and push the pin inward until it seats into a hole on the lever arm. To release the pin, pull and rotate 90°. The lever arm is then free to release the pressure on the system.

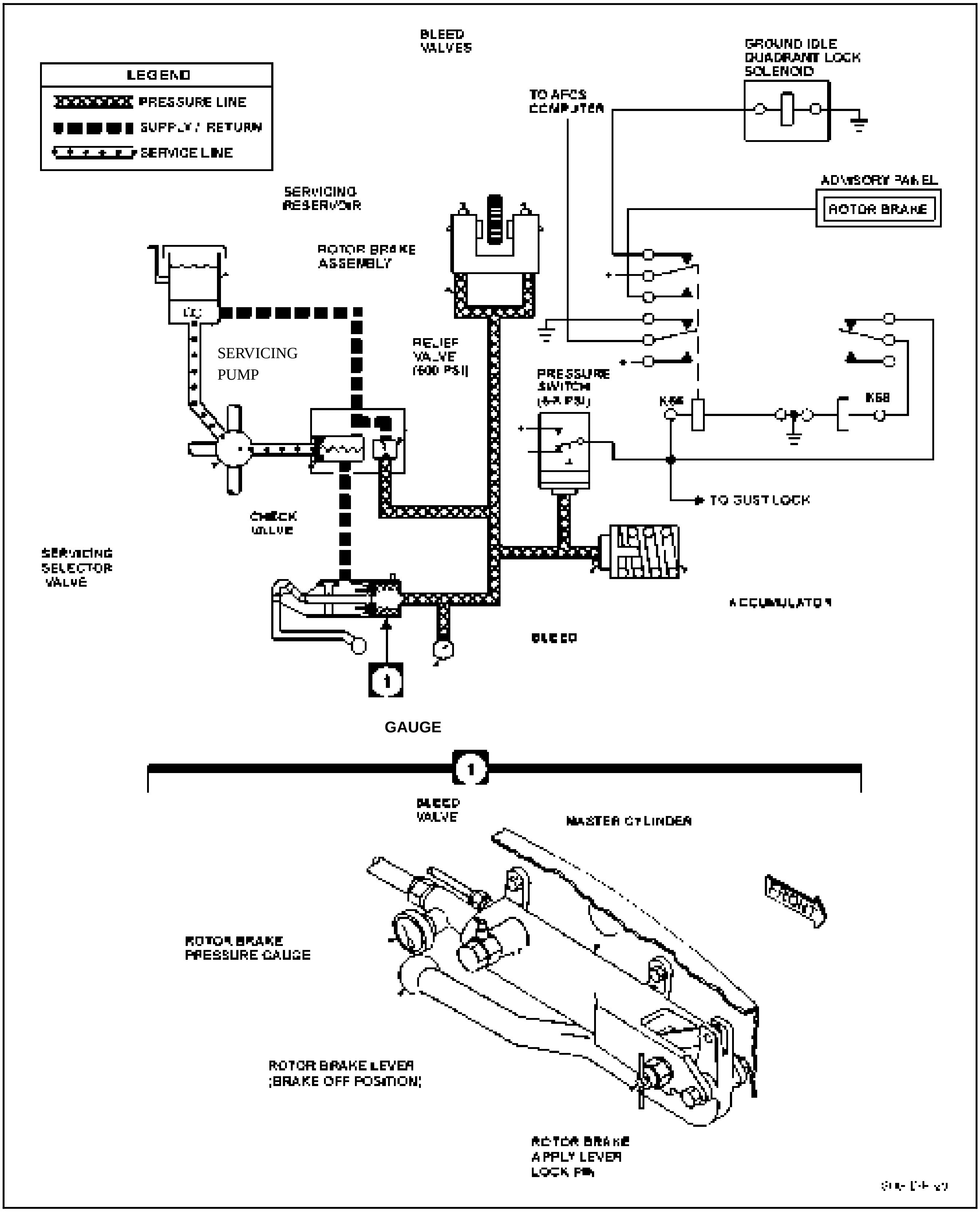


Figure 2-12. Rotor Brake System, Block Diagram

2.2.3.2 Main Rotor Gust Lock

A gust lock is provided as part of the blade indexing unit which is used in conjunction with the automatic main rotor blade-fold sequence.

The primary purpose of the gust lock gear is to index the main rotor head during blade-fold sequences. A secondary purpose is to lock the rotor disk in the spread indexed position. Should the rotor brake hydraulic pressure bleed off in a blade spread condition Figure 2-41, the gust lock may be engaged manually to act as a lock to hold the rotor head in place. This is done by means of the GUST LOCK switch on the miscellaneous switch panel. The GUST LOCK caution light indicates when the gust lock feature of the blade indexing motor has been engaged. The light will illuminate automatically during the blade-fold sequence or whenever the gust lock switch is actuated to the engaged position. The gust lock feature receives power from the DC essential and NO. 2 DC primary buses through circuit breakers marked RTR HD INDEX ENGAGE, on the overhead circuit breaker panel, and BLADE FOLD CONTR, on the SO circuit breaker panel.

CAUTION

Should the rotor brake hydraulic pressure bleed off in a blades folded condition, the gust lock will not prevent the rotor brake disc from turning.

2.3 TRANSMISSION SYSTEM

The primary function of the transmission system is to take the combined power from the two engines, reduce the rpm, and transfer it to the main and tail rotors. The secondary function is to provide a drive for electrical and hydraulic power generation. The powertrain (Figure 2-13) consists of the main transmission modules, drive shaft, an oil cooler, an intermediate gearbox, and a tail gearbox. The tail drive shaft consists of six sections joined by Thomas couplings, with a disconnect coupling at the fold hinge. Thomas couplings between sections eliminate the need for universal joints. The shafts are ballistically tolerant and are suspended at four points in viscous-damped bearings. The oil cooler drive is an integral part of the tail rotor drive shaft system. The intermediate gearbox, located at the base of the pylon, changes angle of drive and reduces tail drive shaft speed. The tail gearbox changes the angle of drive, reduces shaft rpm, and supports and drives the tail rotor. The intermediate and tail gearbox components are designed to be capable of approximately 60 minutes of operation without oil. All other transmission components are designed to be capable of approximately 30 minutes of operation without oil.

2.3.1 Main Transmission

The main gearbox drives and supports the main rotor. The main gearbox is of modular design and has a built-in 3° forward tilt.

The main transmission consists of five modules: two accessory modules, two input modules, and a main module. The left-hand input and accessory modules are identical to the right-hand modules and are interchangeable. A rotor brake is mounted on the tail takeoff, which provides the capability of stopping the rotor system. The rotor brake disc is toothed to provide the means for positioning the main rotor head for blade folding. The main gearbox is pressure lubricated and has oil pressure, oil temperature, low pressure warning, high temperature warning, and chip detector indicating systems incorporated.

2.3.1.1 Input Module

The input modules are mounted on the left and right front of the main module and support the front of the engines. They each contain an input bevel-pinion and gear, and a freewheel unit. The freewheeling unit allows engine disengagement during autorotations. In the case of an inoperative engine, the freewheeling unit allows the accessory module to continue to be driven by the main transmission. The input module provides the first gear reduction between engine and main module.

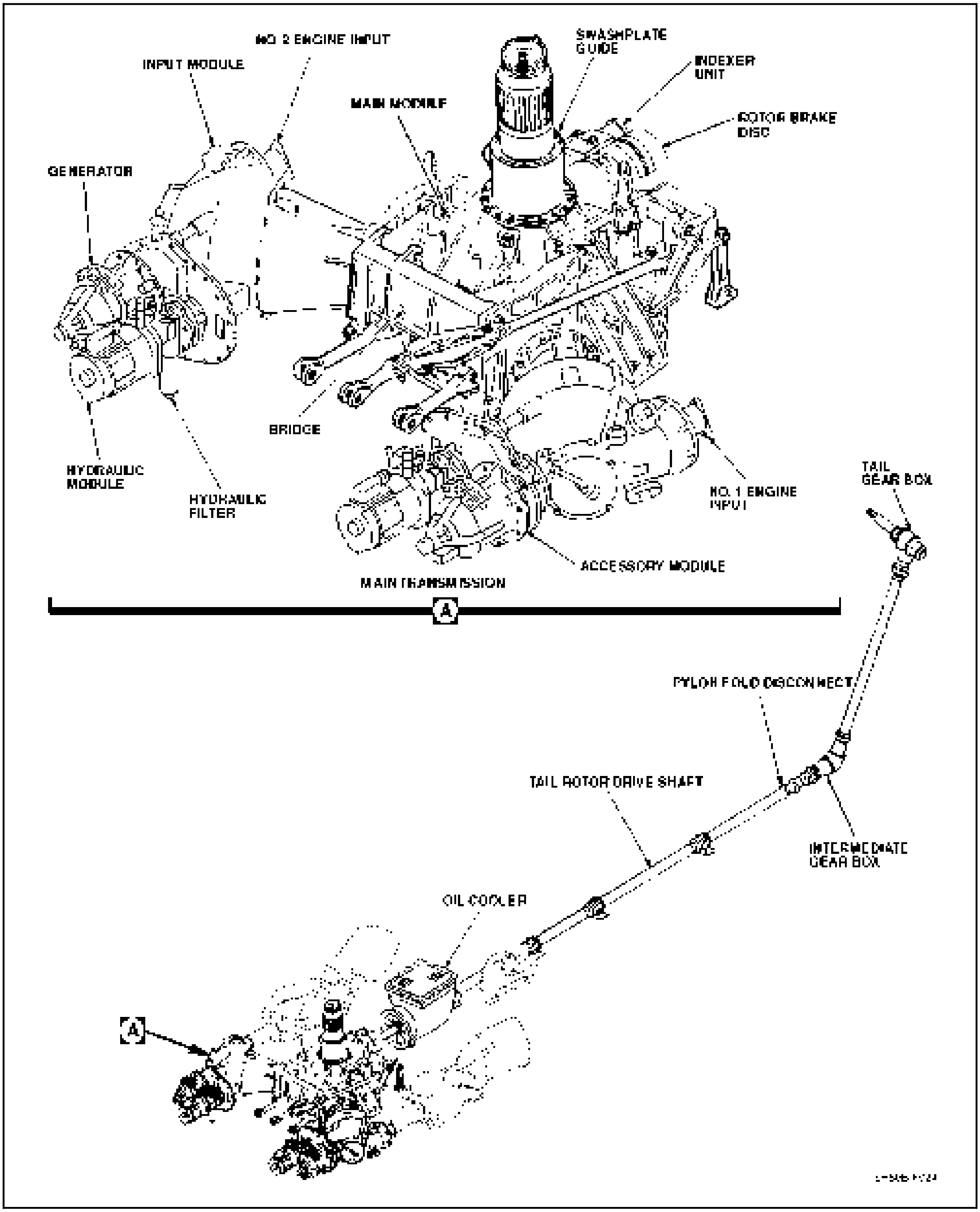


Figure 2-13. Powertrain (Sheet 1 of 2)

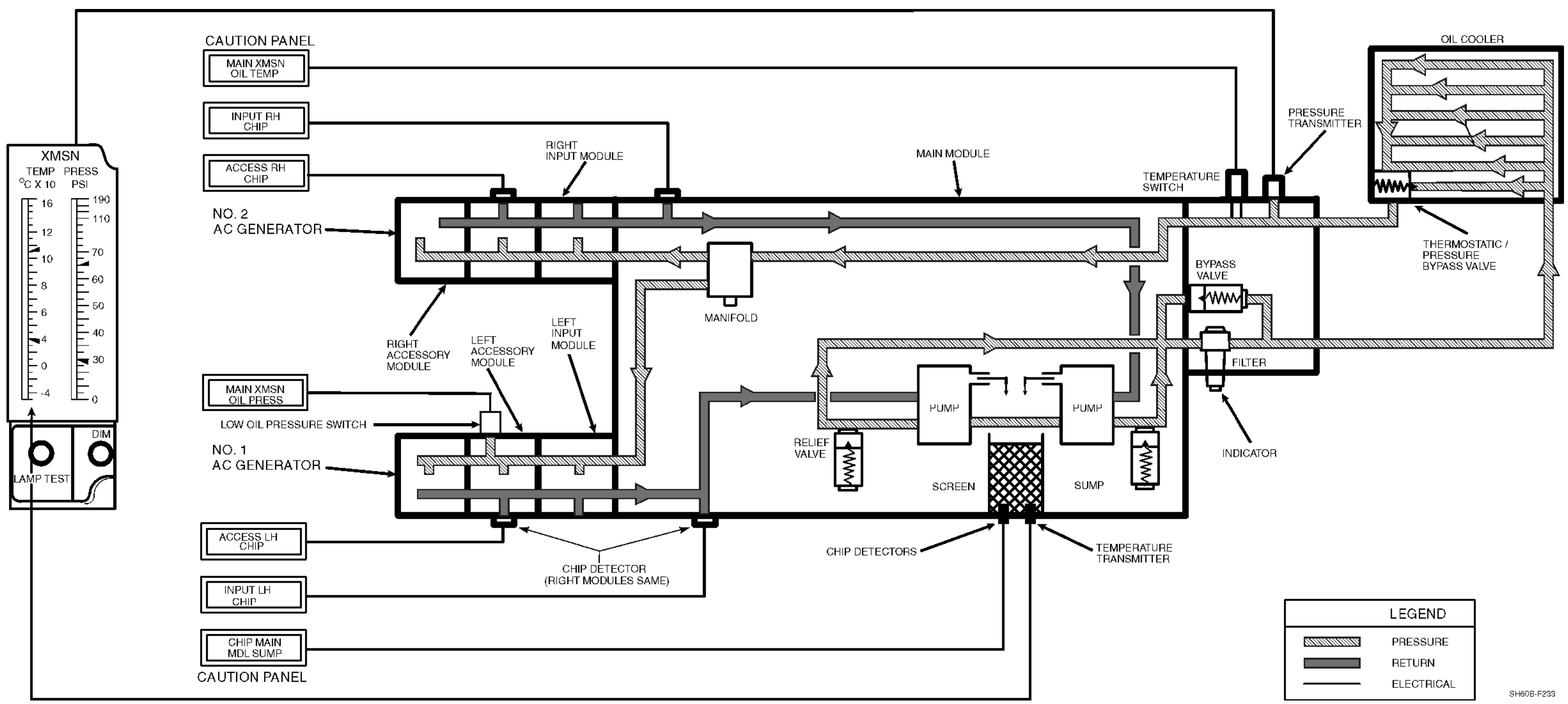


Figure 2-13 Powertrain (Sheet 2)

2.3.1.2 Diaphragm Coupling and Engine Output Shaft

The engine output shaft provides drive from the engine to the input module via the diaphragm coupling. The diaphragm coupling is designed to allow for slight angular or axial misalignment of the engine output shaft during operation.

CAUTION

If an abnormal or loud whining noise is heard during engine startup, shut down engine immediately due to impending diaphragm coupling failure. Maintenance action is required prior to subsequent engine start.

2.3.1.3 Accessory Module

One accessory module is mounted on the forward section of each input module. Each accessory module provides mounting and drive for an AC electrical generator and a hydraulic pump package. A rotor speed sensor is mounted on the right accessory module and supplies rotor speed information to the VIDS. In aircraft with the helicopter emergency egress lighting system (HEELS) system, a left–hand sensor is incorporated which provides N_r information to this system. Additionally, the low oil pressure sensor is mounted on the left accessory module. The accessory modules are always driven by the main transmission.

2.3.2 Intermediate Gearbox

Mounted at the base of the tail pylon is the splash–lubricated, intermediate gearbox (Figure 2-13). It transmits torque and reduces shaft speed from the main gearbox to the tail gearbox.

2.3.3 Tail Gearbox

The splash–lubricated tail gearbox (Figure 2-13) is located at the top of the tail pylon and transmits torque to the tail rotor head. The gearbox mounts the tail rotor, changes the angle of drive, and provides gear reduction. It also enables pitch changes of the tail rotor blades through the flight control system.

2.3.4 Main Transmission Lubrication System

The transmission incorporates an integral wet sump lubrication system (Figure 2-13) that provides cooled, filtered oil to all bearings and gears. Oil is supplied to the hydraulic pump drive shaft and the AC generators for cooling and lubrication. Oil under pressure is supplied through internally cored oil lines, except for the pressure and return lines in and out of the oil cooler. The lubrication system includes two lubrication pumps that are combination pressure and scavenge types operating in parallel. Pressure–regulating and bypass valves protect the lubrication system by returning excess high–pressure oil back to the inlet side of the pump. A two–stage oil filter and various strainers in the sump prevent contamination. The oil filter has a visual impending bypass indicator (red button) that protrudes when the first–stage filter becomes contaminated. When the button pops, the filter element must be replaced to reset. A thermal lockout prevents button popping when oil is cold and thick. The oil cooler uses a blower driven by the tail rotor drive shaft to cool oil before it enters the various modules. The oil cooler has a thermostatic bypass valve that directs oil flow around the oil cooler when the oil temperature is below approximately 54 °C, or if the oil cooler becomes clogged. Other warning and monitoring systems on the main transmission are MAIN XMSN OIL TEMP and PRESS caution lights and XMSN TEMP and PRESS oil gauges. An oil pressure sensor on the left accessory module, the farthest point from the pumps, causes the MAIN XMSN OIL PRESS caution light to illuminate when pressure drops to 14 ±2 psi. The transmission oil temperature warning system is triggered by an oil temperature sensor at the oil cooler input to the main module, located near the tail takeoff drive shaft flange. A caution light marked MAIN XMSN OIL TEMP goes on when transmission oil temperature reaches 117 ±4 °C. Temperature for the gauge is sensed between the sump and the pump. Pressure readings for the gauge are taken at the main module manifold. Electrical power for the warning systems, except chip detection, is from the NO. 2 DC primary bus through the MAIN XMSN circuit breaker on the ATO circuit breaker panel.

2.3.5 Transmission Gauges

The main transmission gauge is located on the CDU as part of the VIDS system and is divided into oil temperature and pressure. The temperature gauge is nonlinear and reads from –50 to 170 °C. An index defines the normal operating range. The pressure gauge is nonlinear and reads from 0 to 190 psi. An index defines the normal operating range. The transmission gauges are powered by the NO. 1 and NO. 2 DC primary buses through circuit breakers, marked NO. 1 and NO. 2 DC INST and located on the ATO circuit breaker panel, and by the NO. 1 and NO. 2 AC primary buses through circuit breakers marked NO. 1 AC INST and NO. 2 AC INST and located on the center and corner circuit breaker panels, respectively.

2.3.6 Transmission Chip Detector System

The transmission chip detector system (Figure 2-14) consists of fuzz–suppression chip detectors and caution lights, marked INPUT LH CHIP, INPUT RH CHIP, ACCESS LH CHIP, ACCESS RH CHIP, and CHIP MAIN MDL SUMP. Five chip detectors provide warning of chips in any of five areas of the main transmission system. Detectors in each module are wired, in parallel, to constantly monitor for metal contamination. Each chip detector can be removed for visual inspection without a loss of oil. A fuzz burnoff feature eliminates false warning due to fuzz and small particles. When a chip is detected and will not burn off, the metal particle triggers the detection system, and a caution light will illuminate. The fuzz burnoff feature will be deactivated when the gearbox oil temperature is above 140 °C to prevent electrical arcing with oil vapor in the gearbox; however, magnetic detection will remain in the main, input, and accessory modules. The main module sump chip detector will turn on the CHIP MAIN MDL SUMP caution light to warn of chips in the main module. The magnetic plugs of the chip detector system will attract ferrous metal chips at any of the detector locations. Should the chip be washed away from the detector, the light will extinguish. The chip detector for the main module sump rests in the lowest point of the oil system and incorporates a 30–second time delay circuit. The accessory module chip detectors are located at the lowest point on the modules themselves, whereas the input module chip detectors are located on the bottom of the main module adjacent to the input modules. The system is powered by the DC essential bus through a circuit breaker on the overhead console circuit breaker panel, marked CHIP DETR.

2.3.7 Intermediate and Tail Gearbox Chip/Temperature Systems

The intermediate and tail gearboxes contain identical chip/temperature sensors that indicate when the gearbox temperature is too high or a chip is present (Figure 2-14). The chip detectors incorporate a fuzz burnoff feature that eliminates false warning due to fuzz and small particles. The fuzz burnoff feature will be deactivated when the gearbox oil temperature light is lighted; however, magnetic detection will remain to light the caution light. When a chip is detected and will not burn off, a caution indicator on the caution/advisory panel will light, indicating CHIP INT XMSN or CHIP TAIL XMSN. The oil temperature sensor is a bimetal strip that reacts to temperatures. When the oil temperature reaches 140 °C, a switch closes to turn on a caution light in the cockpit, marked INT XMSN OIL TEMP or TAIL XMSN OIL TEMP. Power to operate the chip system is provided from the DC essential bus through a circuit breaker marked CHIP DETR. Power to operate the oil temperature system is from the NO. 2 DC primary bus through a circuit breaker marked MAIN XMSN and located on the ATO circuit breaker panel.

2.3.8 Chip-Detector Caution Lights Self-Test

All the transmission modules and the intermediate gearbox and tail rotor gearbox chip detectors have self–test circuits for the caution lights. The test circuit is activated when the caution/advisory panel BRT/DIM, TEST is released after the caution/advisory panel test sequence. The self–test checks for both short and open circuit faults. If a short or open circuit is present, the appropriate chip caution light will flash at approximately 2 flashes per second for a total of 16 flashes, and the master caution light will remain illuminated. If there is no fault when the test switch is released, only the master caution capsule will flash.

Note

The self–test checks circuitry up through the fuzz burnoff module and does not check the ability of the detector to detect chips or if a detector is installed.

2.3.9 Main Gearbox Vibrations

The main gearbox contains many possible sources of high–frequency vibrations, such as the various gearbox–mounted accessories, the accessory gear train, oil–cooler blower, and the input–bevel gear and freewheeling units. These vibrations are generally heard rather than felt. Combinations of these high–frequency vibrations in extreme cases could result in the pilot sensing low– or medium–frequency vibrations. These would be detected as vibrations which are affected only by variation in main rotor speed and may be just as apparent in a ground run as in flight. There are also numerous gear clash sounds that occur under various conditions, the acceptability of which can only be determined by experience or measurements with instrumentation.

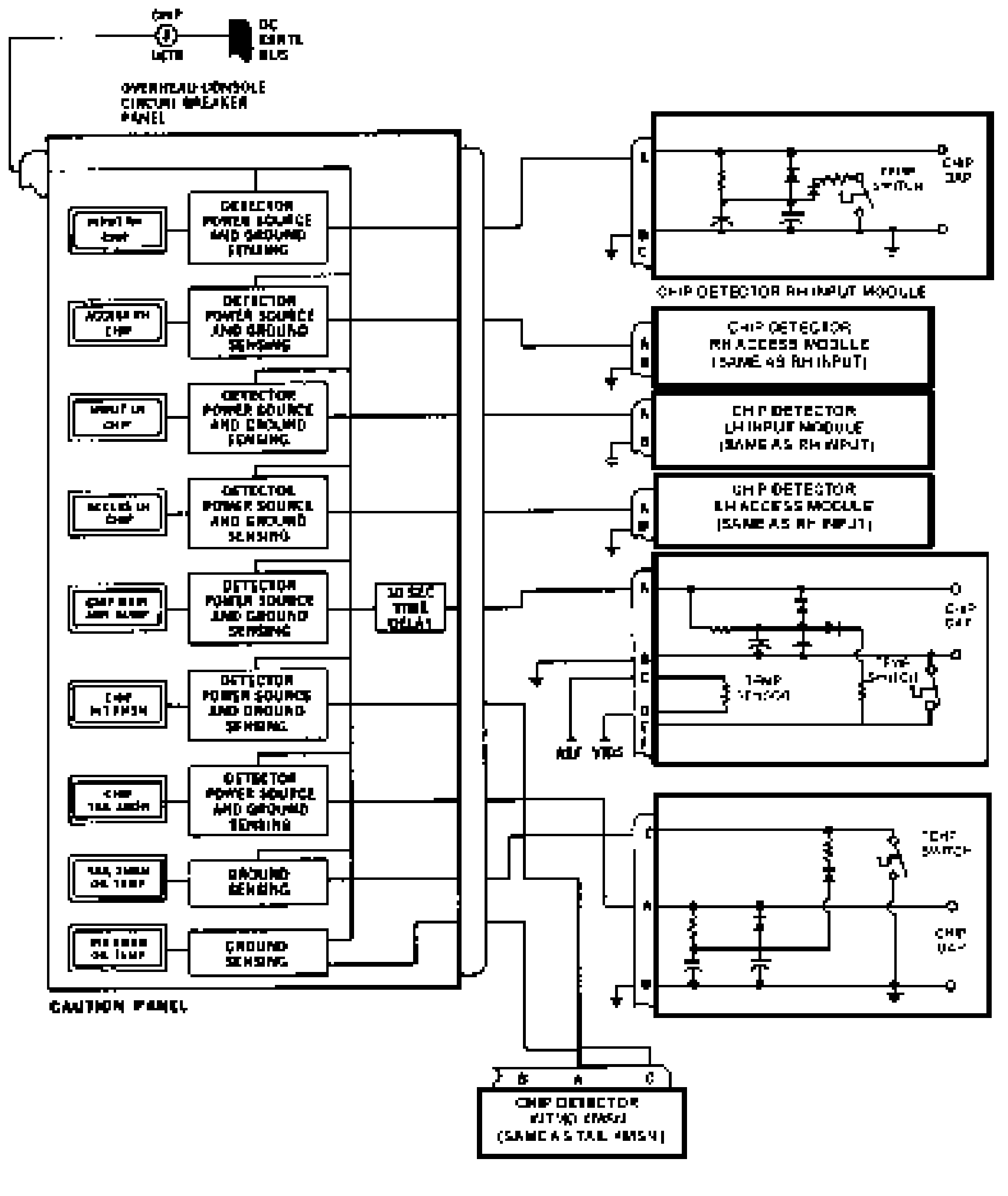


Figure 2-14. Powertrain Chip Detector System

2.4 FUEL SYSTEM

The fuel supply system (Figure 2-15 prior to BuNo 162349, Figure 2-16 BuNo 162349 and subsequent) is a crashworthy, suction-type system consisting of two internal main cells interconnected to form a single tank, a fuel line network, firewall-mounted selector valves, prime/boost pump, engine-driven boost pumps, and engine fuel filters. The left internal cell has provisions for single-point refuel/defuel, gravity refuel, and the helicopter in-flight refueling (HIFR) system. It also contains two high-level shutoffs, two check valves, sump drain, and vent. The right internal cell contains two check valves, a sump drain, a vent, an APU fuel line, and the fuel jettison system. Total system capacity (usable) is 590 gallons internal. Additionally, BuNo 162349 and subsequent incorporate an auxiliary fuel system capable of supporting two external auxiliary fuel tanks containing a total of 240 gallons of fuel (120 gallons in each auxiliary tank).

2.4.1 Main Fuel Supply Operation

The prime/boost pump primes all fuel lines if prime is lost and also acts as an APU boost for APU starts and operations. A selector valve, driven by a cable from the fuel selector lever, permits the operation of either engine from either cell. All lines are routed in the most direct manner and include self-sealing breakaway valves that stop fuel flow in the event of fuel system damage. Fuel from both cells is drawn by suction to the engine-driven boost pump, then pumped through the engine fuel filter to the HMU high-pressure pump.

The engine fuel pressure warning system for each engine consists of a pressure switch that illuminates the caution lights, marked #1 or #2 FUEL PRESS, when fuel pressure drops below 8 to 10 psi from the respective engine-driven boost pump. This visually indicates a possible malfunction in the engine-driven fuel boost pump or an air leak in the fuel system.

The engine fuel-filter bypass warning system for each engine consists of an electrical switch, impending bypass popout button (located on the filter), and caution lights. Once the filter goes into bypass, the caution lights, marked #1 or #2 FUEL FLTR BYPASS, will light.

Note

The fuel filters are not sensitive to water contamination. Water-contaminated fuel may cause fluctuations/surges in one or both engines with no associated FUEL PRESS or FUEL FLTR BYPASS caution light.

The #1 and #2 FUEL FLTR BYPASS and the #1 and #2 FUEL PRESS caution lights are powered by the NO. 1 and NO. 2 primary DC buses, respectively, through circuit breakers marked NO. 1 ENG and NO. 2 ENG WARN LTS located on the ATO circuit breaker panel.

2.4.1.1 Fuel Selector Levers

There are two ENG FUEL SYS levers, one for each engine, located outboard of the ENG POWER CONT levers (Figure 1-6). The fuel selector levers manually position the fuel selector valves to any one of three positions: OFF, DIR, or XFD. The fuel selectors are connected to the fuel selector valves with low-friction, flexible push-pull cables. With the selectors at OFF, the fuel selector valves are closed, allowing no fuel to the engines. When the selectors are moved forward to DIR, the fuel selector valves are opened, providing fuel flow for each engine from its individual cell. Moving the selector to XFD provides fuel to the engine from the opposite cell through the crossfeed system. A check valve in each crossfeed line prevents air from the fuel line of an inoperative engine from crossing to the operating one. When either fire emergency control T-handle, located outboard of the fuel selector levers, is pulled aft, the respective fuel selector lever will be mechanically placed in the OFF position.

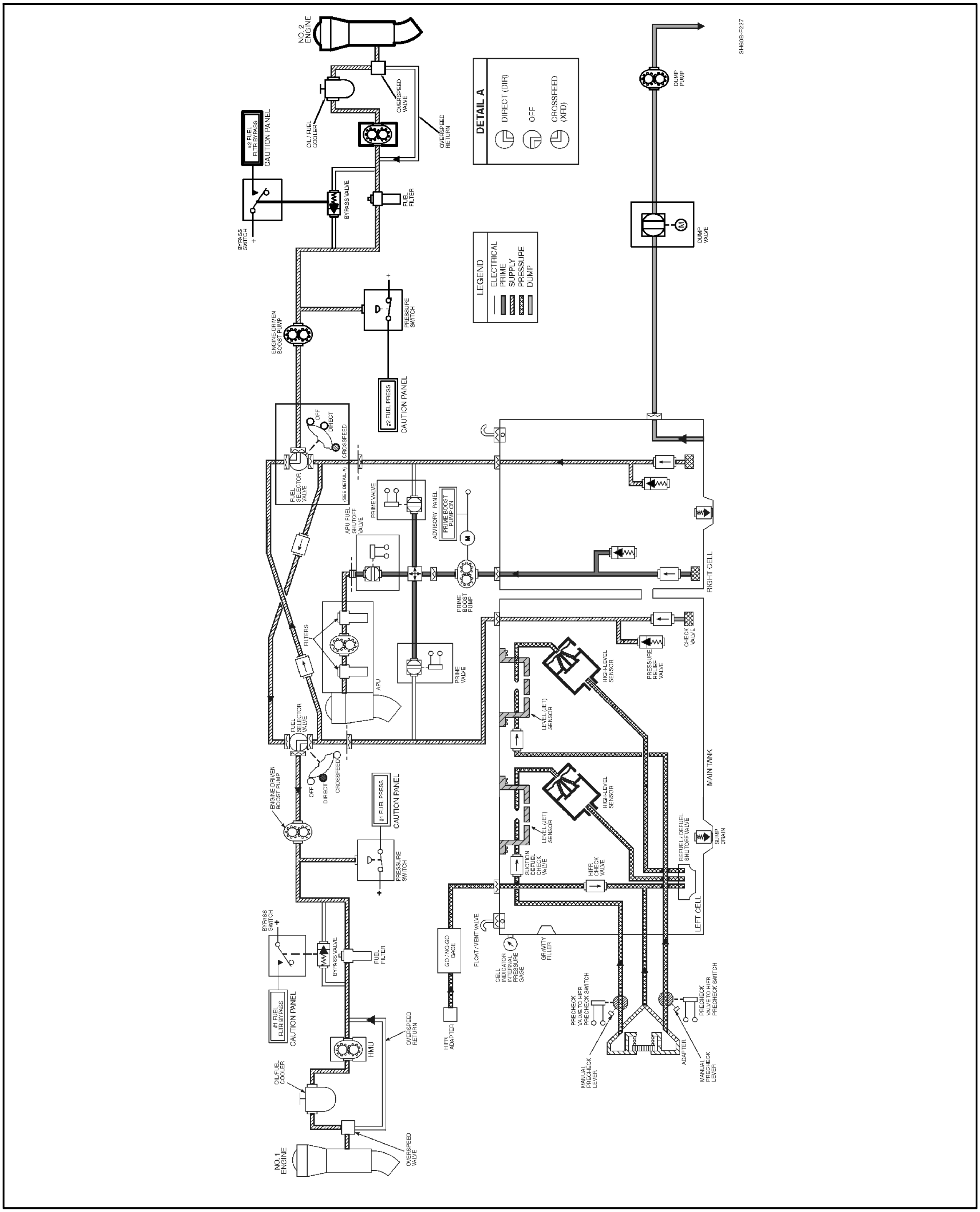


Figure 2-15. Fuel System Block Diagram — Prior to BuNo 162349

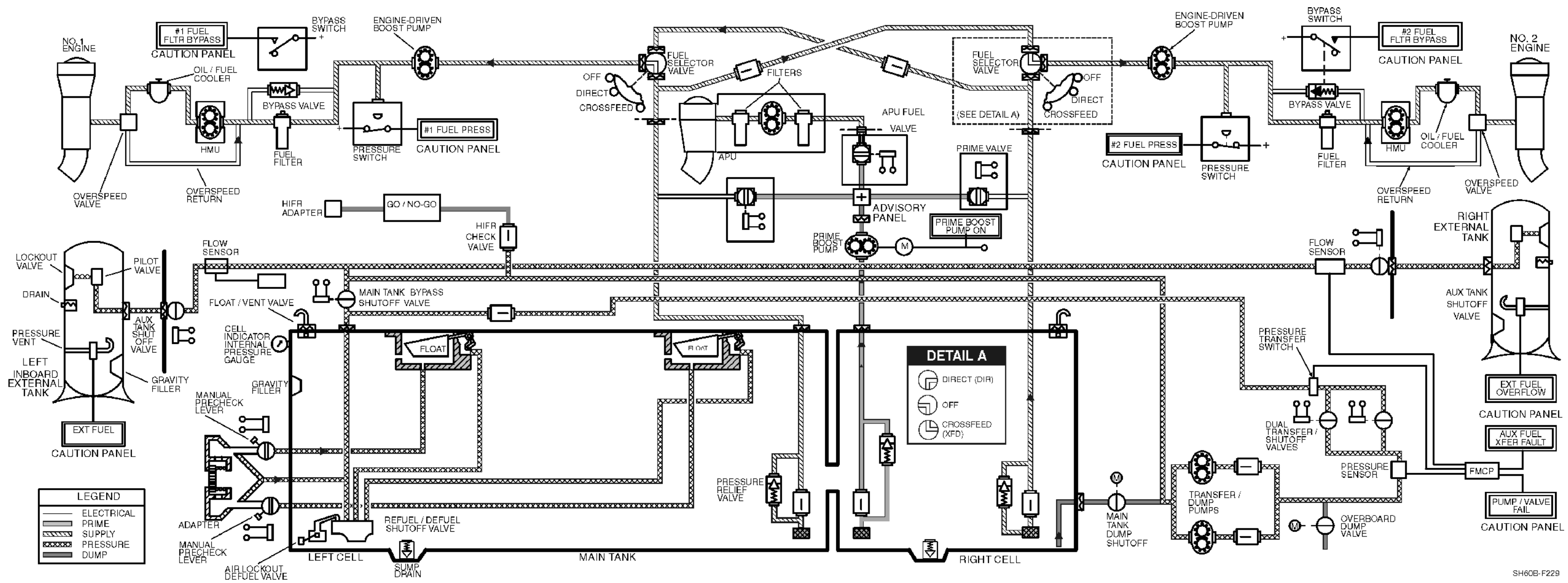


Figure 2-16. Fuel System Block Diagram — BuNo 162349 and Subsequent

2.4.1.2 Engine/APU Boost/Fuel Prime System

The APU boost and engine prime system consists of a suction line located in the right fuel cell, a prime/boost pump (which is externally mounted on the tank), APU fuel shutoff valve, two engine fuel prime shutoff valves, and a selector switch located on the overhead console (Figure 1-6). The FUEL PUMP selector switch is a three-position switch, marked APU BOOST, OFF, and FUEL PRIME. Activation of the switch illuminates the PRIME BOOST PUMP ON advisory light. The FUEL PRIME position allows fuel to enter all fuel lines before engine start. Power to operate the prime-boost pump is from the battery bus through a circuit breaker marked FUEL PRIME BOOST and located on the center console circuit breaker panel.

2.4.1.2.1 Fuel Prime System Operation

Placing the APU BOOST/FUEL PRIME switch to APU BOOST opens the APU fuel shutoff valve and activates the fuel prime/boost pump. Placing the APU BOOST/FUEL PRIME switch to FUEL PRIME opens both engine prime shutoff valves and allows individual priming of the engines with the PCL in lockout.

2.4.2 Main Tanks Fuel Quantity System

The fuel quantity system (Figure 2-17) consists of a fuel probe mounted in each fuel cell and a fuel quantity signal conditioner. The system interconnects two SDCs to the VIDS CDU.

2.4.2.1 Fuel Low Level Warning System

The fuel low level warning system consists of a dual-channel, low-level warning conditioner; two fuel cell-sensing units; and associated caution lights. Each cell-sensing unit is placed at approximately the 200 to 225 pound fuel level. As long as the sensing unit is covered with fuel, the warning conditioner will keep the #1 and #2 FUEL LOW caution lights off. As fuel is consumed and its level drops below the 200 to 225 pound level, the fuel cell-sensing units sense the lack of fuel and signal the conditioner of a low fuel state. These lights will continue to illuminate and extinguish as long as fuel continues to wash on and off the sensors. This system is completely independent of the fuel quantity system. The conditioner then applies power to the appropriate FUEL LOW caution and MASTER CAUTION lights. Power to operate the fuel low level system warning is from the NO. 1 DC primary bus through a circuit breaker marked FUEL LOW WARN and located on the ATO circuit breaker panel.

2.4.3 Fuel Dump System

A fuel dump system (Figure 2-15 prior to BuNo 162349, Figure 2-16 BuNo 162349 and subsequent) is installed to allow for emergency rapid dumping of fuel at approximately 836 pounds per minute, but can exceed 1,000 pounds per minute. The system consists of a FUEL DUMP switch mounted on the lower console EMER panel and a standpipe and dump port in the right main fuel cell. Aircraft prior to BuNo 162349 have a dump valve and a pump/motor assembly in the transition section. Aircraft BuNo 162349 and subsequent have a main tank shutoff valve, an overboard dump shutoff valve, two transfer/dump pumps/motors, and Fuel Management Control Panel logic to control required valves. Power to operate the fuel dump system is from the DC essential bus through a circuit breaker marked FUEL DUMP CONTR, the NO. 1 AC primary bus through a circuit breaker marked FUEL DUMP PUMP, and, for BuNo 162349 and subsequent, the NO. 2 AC primary bus through a circuit breaker marked FUEL DUMP PUMP.

The circuit breakers are located on the overhead circuit breaker panel and the center circuit breaker panel, respectively, and, for BuNo 162349 and subsequent, the corner circuit breaker panel.

WARNING

After dumping, observe the fuel readout to ensure dumping has ceased.

Note

Fuel can be dumped when the helicopter is on the ground. The fuel dump system is not protected by the weight-on-wheels (WOW) switch.

2.4.3.1 System Operation

Placing the FUEL DUMP switch to the DUMP position opens the valve and closes the contacts in the pump/motor relay, operating the pump/motor. Fuel will dump out of both cells down to the level of the cell interconnect. Fuel will then continue to dump out of the right cell to the level of the standpipe. After total dump, approximately 270 pounds will remain in the left cell and 140 pounds in the right cell (prior to BuNo 162349). Fuel quantity remaining after dump can vary depending on aircraft pitch attitude.

2.4.3.2 System Operation (with FMCP)

With the FMCP MASTER and MODE switches set to TRANSFER and MANUAL OVERRIDE, placing the FUEL DUMP switch on the emergency control panel (Figure 1-7) to the DUMP position signals the FMCP, bypassing all of the FMCP operating modes. The FUEL DUMP indicator light on the FMCP illuminates, the auxiliary tank valves open, both transfer/shutoff valves close, the main tank bypass shutoff valve closes, the overboard dump valve opens, and both fuel transfer/dump pumps begin running. When the auxiliary tanks are empty, each auxiliary tank valve closes and the main tank dump valve opens.

The fuel transfer/dump pumps continue running and fuel will dump out of both main cells down to the level of the interconnect (approximately 270 pounds in each cell). Fuel will then continue to dump out of the right cell to the level of the low level sensor (approximately 210 pounds) when the dump signal is interrupted at the FMCP (BuNo 162349 and subsequent). The fuel quantity remaining in the right cell after dump could be lower at noseup attitudes associated with lower airspeeds. When the dump signal is interrupted, the main tank dump valve closes, the main tank bypass shutoff valve opens, and both fuel transfer/dump pumps stop. Placing the FUEL DUMP switch to OFF closes the overboard dump valve and restores FMCP control functions.

Note

- Regardless of FMCP switch positions or whether auxiliary tanks are installed, selecting Fuel Dump on the Emergency Control Panel will enable emergency dumping from the main tank. FMCP switches need only be set to TRANSFER and MANUAL OVERRIDE if pilots desire to dump fuel from the auxiliary tanks. With no auxiliary tanks installed, fuel may be dumped from the main tank with the FMCP in any configuration, including STOP FLOW.
- For aircraft operating without a functional FMCP, refer to paragraph 2.4.3.1.

2.4.4 Single-Point Pressure Refueling System

The single-point pressure refueling system is used to pressure refuel the fuel tanks on the ground. The fuel tanks may be serviced without electrical power. On aircraft BuNo 162349 and subsequent, without power applied to the aircraft, all tank valves are open and fuel enters all tanks randomly until the main tank high level sensor is reached and the mechanical shutoff float valve closes. During refueling with electrical power on, all auxiliary tank shutoff valves are closed and the main fuel tank is filled first. When the FMCP is signaled that the main tank is filled, the right inboard auxiliary and left inboard auxiliary tanks, if sensed as installed, are then filled by the same process in order.

CAUTION

During single-point pressure refueling, if the right cell fills faster than the left cell, monitor the fuel quantity gauges closely. If the difference in cell quantities persists, stop refueling before the right cell is full (approximately 1,700 pounds).

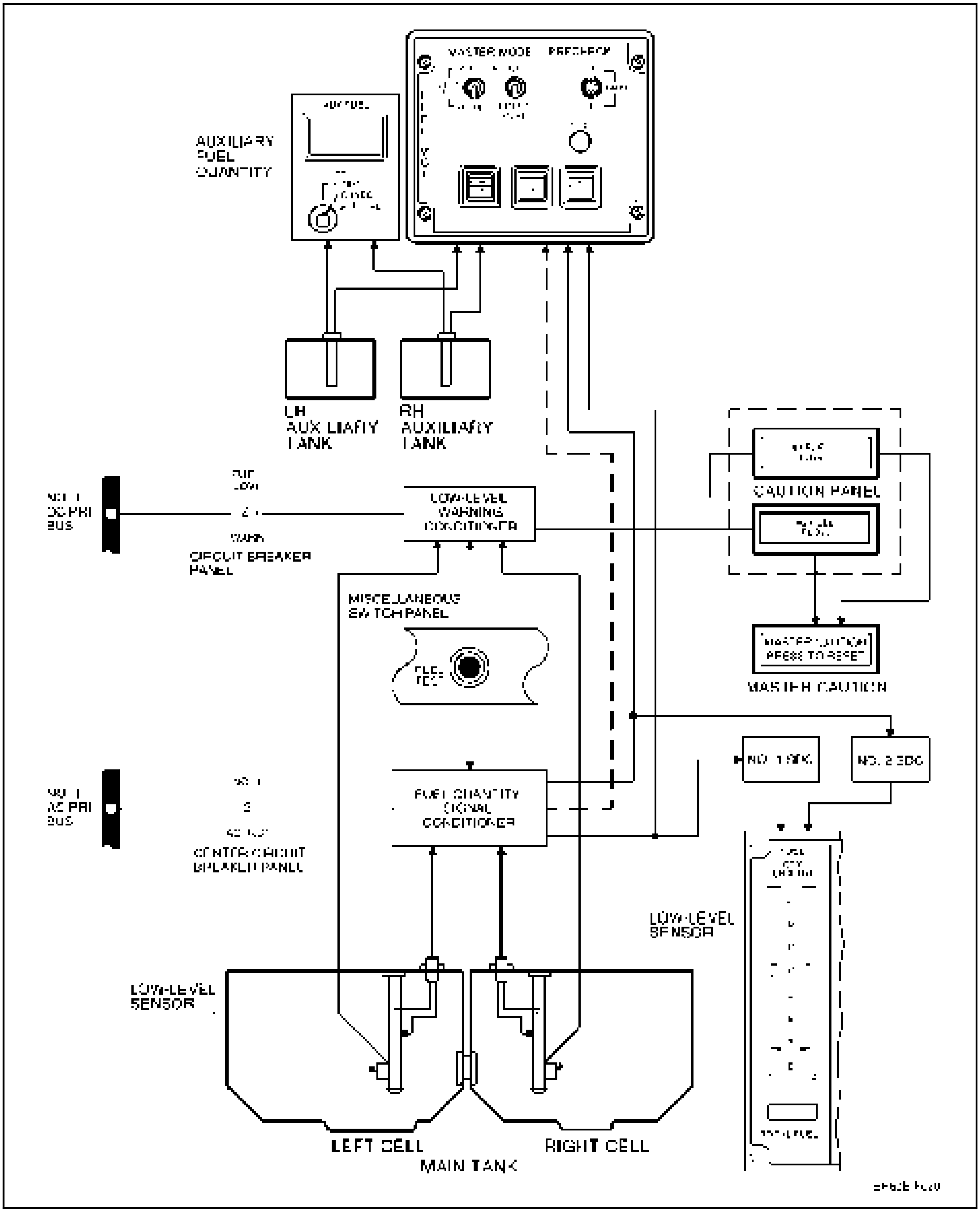


Figure 2-17. Fuel Quantity System (with Auxiliary Tanks Installed)

Note

Fuel quantity indicators are not operable without AC electrical power and fuel quantity must be visually checked.

2.4.4.1 Pressure Refueling Panel

The pressure refueling panel (Figure 1-3, index NO. 34), located on the left side of the aft fuselage, provides a single point for refueling and defueling. The pressure refueling panel contains a connecting adapter, pressure gauge, and two manually operated precheck valves. Aircraft prior to BuNo 162349 have two jet sensors in the left fuel cell that will activate their respective shutoff valves when fuel in the cell immerses them. Either or both shutoff valves will in turn signal the pressure refuel/defuel valve to interrupt normal fueling.

The two high-level sensors in the left cell will cause fuel flow to be reduced to 5 gallons per minute when the cells are becoming full. When the precheck valves are pressed, jet sensor immersion is simulated and fuel flow is interrupted to indicate that the system is operating properly. Aircraft BuNo 162349 and subsequent incorporate two high-level float sensors in the top of the tank that will stop fuel flow when the main tank is full. When the precheck valves are pressed, fuel is redirected to raise the floats, which then stops the fuel flow to indicate that the system is operating properly. In all aircraft, the high-level sensors can also be tested during a HIFR from inside the aircraft. Pressure that may have built up in the tank, due to a clogged or malfunctioning vent, will register on the tank internal pressure gauge. Specific instructions for conducting a pressure refueling precheck are on a decal below the pressure refueling adapter and in the servicing chapter (Chapter 3).

2.4.4.2 Gravity Refueling

Gravity fueling ports are available for the main and external auxiliary fuel tanks. Tanks may be gravity fueled in any order.

2.4.4.3 Suction Defuel

All tanks can be suction defueled from the pressure refueling port, except the 270 pounds in the right main cell below the tank interconnect, which must be suction defueled through the cell sump drain valve.

2.4.4.4 HIFR Refueling System

The HIFR system (Figure 2-18) consists of a Wiggins quick-disconnect pressure-refueling fitting, a pressure-refueling precheck switch, and a five-element (fuse) GO/NO-GO canister to pressure refuel the main fuel tanks. The Wiggins fitting is located above the right-hand fuel cell just forward of the GO/NO-GO canister. The GO/NO-GO canister is mounted above the right fuel tank and permits only acceptable fuel to pass. The elements are water sensitive and will shut off fuel flow at a 20 psi differential pressure. Flow is reduced to an extremely low level if the fuel is contaminated with water and particulate matter above a predetermined level.

Note

If the helicopter must be fueled when the quality of the fuel is in question, it should be refueled through the HIFR fitting. The HIFR filter is capable of removing both water and particulate matter from fuel.

Fuel spillage is collected in a drip pan located on top of the fuel tank and then drains overboard. The precheck panel contains a ground connector and a precheck switch. When the switch is moved to PRECHECK, 28 Vdc power is applied through the switch to the pressure refuel precheck valve in the refueling line to shut off fuel flow to the high-level sensor, testing the complete refueling system. Power to operate the precheck system is from the NO. 1 DC primary bus through a circuit breaker marked HIFR TEST and located on the SO circuit breaker panel.

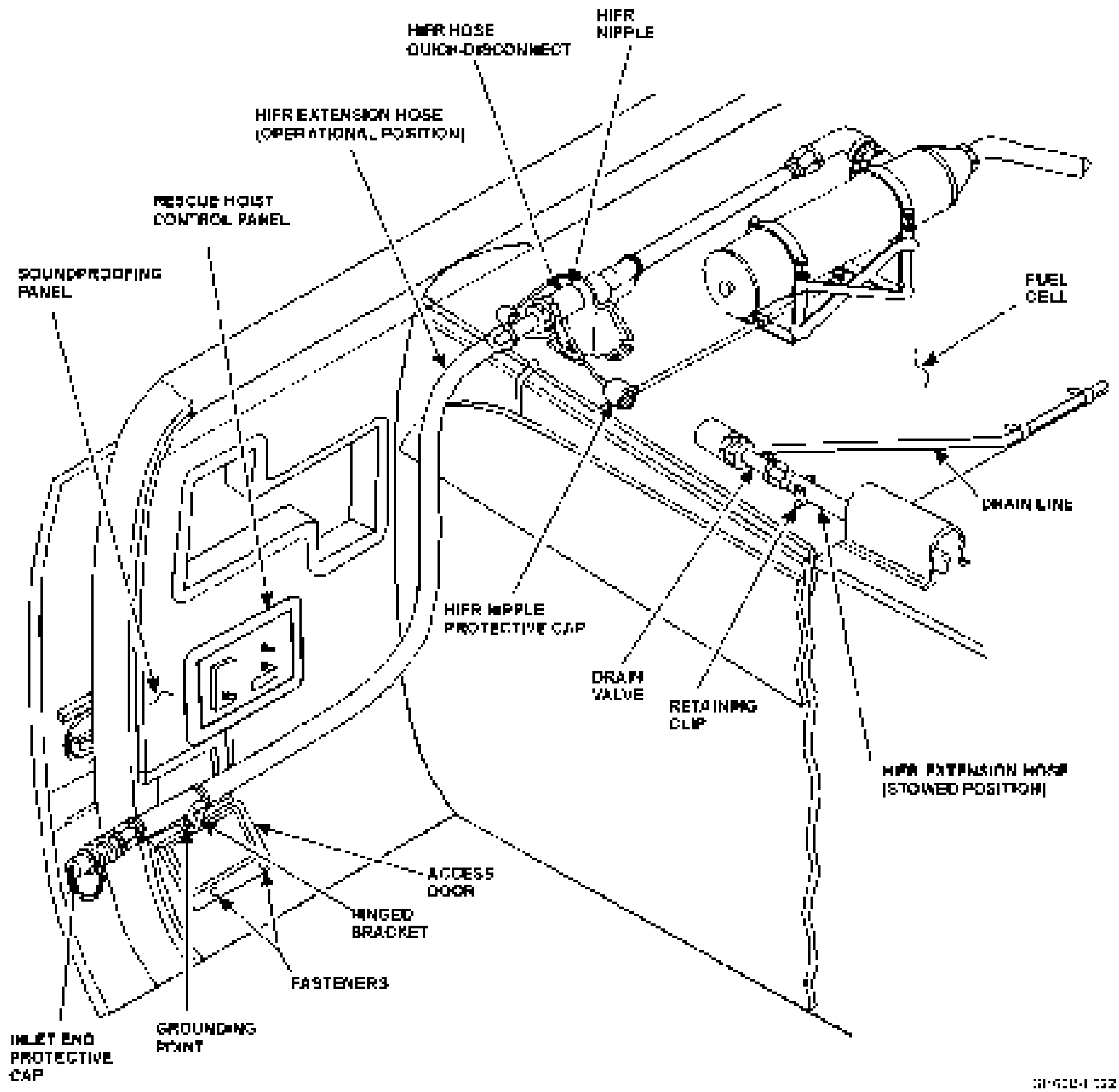


Figure 2-18. Helicopter In–Flight Refueling (HIFR) System (Sheet 1 of 2)

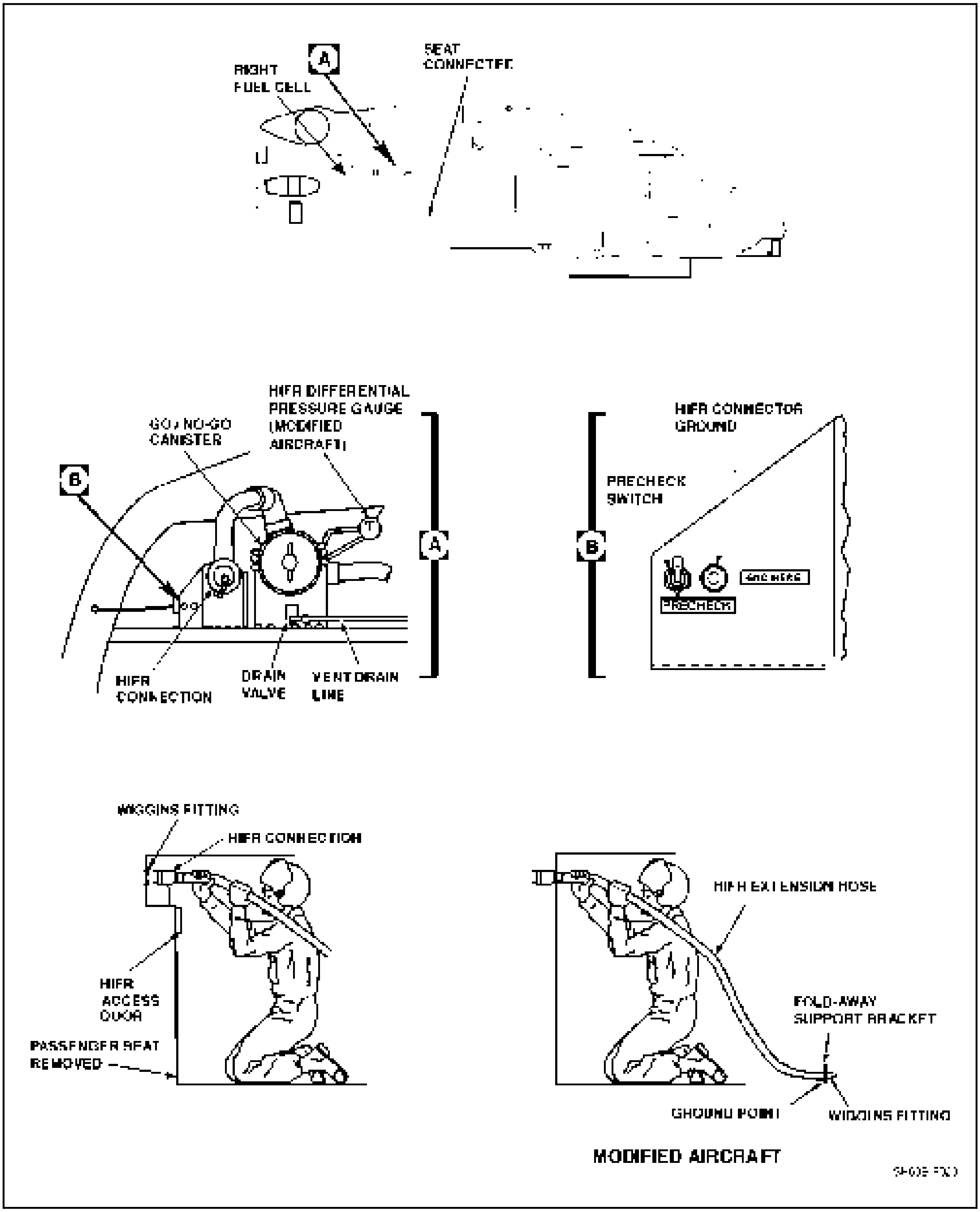


Figure 2-18 Helicopter In–Flight Refueling (HIFR) System (Sheet 2)

Some aircraft are modified to include a HIFR extension hose and a differential pressure gauge. The extension hose is stowed on top of the fuel cell and held in place with a retaining clip. For HIFR, the extension hose is connected to the HIFR connection, routed, and secured to a foldaway support bracket located aft of the personnel door below the rescue hoist control panel. The refueling hose is then connected to the extension hose Wiggins fitting at the support bracket. A grounding jack is located on the foldaway support bracket. The differential pressure gauge shows HIFR GO/NO–GO canister input and output pressure differential. A drain valve can be manually activated to clear the HIFR system of fuel.

If the left tank vent valve should malfunction and remain stuck in the closed position during HIFR, uneven filling of the main tank cells may occur. Once the fuel level has risen above the interconnect opening (approximately 600 pounds total), trapped air in the left cell will slow the rate at which the left cell fills with fuel to the point that the level will not rise while the right cell will continue to fill normally. If the fuel level remains below the high level shutoff sensor located in the left cell, the fuel flow will not stop. The fuel level in the right cell will rise to a level that will force the vent valve to close, resulting in an overpressure condition and possible cell rupture. See Chapter 8, for HIFR procedure.

CAUTION

During HIFR, if the right cell fills faster than the left cell, monitor the fuel quantity gauges closely. If the difference in cell quantities persists, stop refueling before the right cell is full (approximately 1,700 pounds).

2.4.5 Auxiliary Fuel System

The auxiliary fuel system is comprised of two auxiliary tank locations, a fuel management system, and fuel quantity display information. Management is provided through the FMCP (Figure 2-16 and Figure 2-17). The FMCP receives the sensor switch signal and provides logic to the control valves and pumps to control auxiliary fuel system functions.

2.4.5.1 Fuel Management Control Panel

The fuel management control panel (FMCP) functions are software controlled. The FMCP panel marked FUEL MGT is on the center console (Figure 1-7). The MASTER switch is a three–position switch marked TRANSFER, STOP FLOW, and REFUEL. At T

TRANSFER, the FMCP control logic enables the MODE switch and transfer operation. At STOP FLOW, power is removed from all FMCP switches, control logic is disabled, and auxiliary fuel system control is stopped, which closes all auxiliary tank shutoff valves. At REFUEL, when refueling with electrical power on, the aircraft the sequencing logic turns off all FMCP switches except precheck switches, all auxiliary tank shutoff valves are closed until the main internal tank is filled, and the auxiliary tank shutoff valves open one at a time to selectively fill the auxiliary tanks. The fuel system fill sequence is main tank, right inboard auxiliary tank, and left inboard auxiliary tank.

The MODE switch is a two–position switch marked AUTO and MANUAL OVRD. At AUTO, the fuel management logic is not initiated until the main fuel tank fuel level depletes to 2,700 to 2,580 pounds. If two auxiliary tanks are installed, the second tank will transfer after the main tank fuel level again drops enough to accommodate the complete auxiliary tank. When in MANUAL OVRD, auxiliary tank fuel is immediately transferred to the main tank until the high level sensor is reached, or until the auxiliary tanks are empty. Fuel tank transfer sequence is left inboard auxiliary tank then right auxiliary tank.

The PRECHECK switch is marked A, MAIN, and B and is spring loaded to the center (MAIN) position. The center position provides power to the precheck valves in the main fuel tank. When moved to A or B, power to the respective precheck valve is interrupted and fuel flow during refueling is immediately stopped, and fuel flow indication of the FMCP FLOW lights will go out.

Note

A malfunctioning FMCP (with electrical power applied) can cause activation of precheck valves preventing the ability to pressure refuel or HIFR. The FUEL MGMT circuit breakers on the ATO circuit breaker panel must be pulled to secure the precheck valves to allow fueling.

Three fuel flow indicators/selectors are marked L INBD FLOW, FUEL DUMP, and R INBD FLOW. The FUEL DUMP indicator lights when the EMER PNL FUEL DUMP switch is activated. The auxiliary tank indicators are split. L INBD and R INBD show at all times. FLOW lights independently when fuel is sensed flowing into or out of respective auxiliary fuel tank. The L INBD and R INBD selector switch function is a momentary pushbutton which when pressed, and MANUAL OVRD is selected, will initiate appropriate circuits to transfer fuel from an auxiliary tank to the main tank. When fuel level in the main tank reaches the high level shutoff, all manual transfer commands are stopped. When a selected auxiliary tank is emptied before the main tank high level shutoff is reached, the transfer circuits will shut off in 10 seconds.

Power to operate the FMCP is from the NO. 1 and NO. 2 DC primary buses through two circuit breakers marked FUEL MGMT and located on the ATO circuit breaker panel.

2.4.5.2 Fuel Transfer System

The fuel transfer system is controlled by the FMCP. It provides automatic or manual transfer of fuel from the auxiliary fuel tanks to the main fuel cells. Dual transfer/dump pumps suck fuel from the auxiliary fuel tanks through shutoff valves and deliver the fuel to the main tank. Pressure switches are provided as sensors to enable the FMCP to monitor fuel system operation. Fuel transfer is approximately 285 pounds per minute. When one pump fails to transfer fuel, and the second pump successfully transfers fuel, the PUMP/VALVE FAIL caution light illuminates. The FMCP receives fuel quantity status from main and auxiliary systems to properly schedule fuel transfer from auxiliary tanks to the main tank.

CAUTION

- During transfer of auxiliary fuel, if the right cell fills faster than the left cell, monitor the fuel quantity gauges closely. If difference in cell quantities persists, stop transferring fuel before the right cell is full (approximately 1,700 pounds).
- Do not initiate unmonitored manual transfer to the main tanks from auxiliary tanks until main tanks are below 3,200 pounds for an external auxiliary tank transfer. During manual auxiliary tank transfer, the main tank high level sensor (float valves) should prevent overflow of the main fuel tanks.

2.4.5.3 Manual Fuel Transfer Check

When main fuel tank capacity has decreased approximately 300 pounds, check the manual fuel transfer system to ensure proper transfer. Two short manual transfers will exercise both dual transfer pumps and transfer valves to ensure proper transfer.

2.4.5.4 Auxiliary Fuel Tanks

Each inboard weapons pylon is configured to accept a 120 gallon drop tank. The fuel level of each auxiliary fuel tank is internally monitored by a single gauge probe. Each probe provides a fuel quantity signal to the FMCP and the auxiliary fuel quantity indicator. Each auxiliary tank contains a low–level thermistor sensor which is exposed only when the tank is empty and then sends a signal to the FMCP. Each external auxiliary tank contains an overflow thermistor sensor which sends a signal to the FMCP if fuel is sensed in the external tank vent line. When fuel is sensed, a signal is sent to illuminate the EXT FUEL OVERFLOW caution light. Power to operate the auxiliary fuel tanks is from the NO. 2 DC primary bus through two circuit breakers marked FUEL LH INBD and FUEL RH INBD and located on the SO circuit breaker panel.

The auxiliary fuel quantity indicator marked AUX FUEL is on the pilot instrument panel (Figure 1-8). The window marked LBS provides a digital reading of the fuel quantity for the auxiliary tank or tanks selected with the selector switch. A selector switch marked L INBD–R INBD–TOTAL selects the fuel quantity in the left or right auxiliary tank or the total quantity of the two tanks. Power to operate the auxiliary fuel quantity indicator system is from the NO. 2 DC primary bus through a circuit breaker marked FUEL MGMT and located on the ATO circuit breaker panel.

Any combination of auxiliary tanks can be installed or removed from the aircraft. The software logic of the FMCP senses if an auxiliary tank gauge and low-level switch signal is absent. The FMCP commands automatically bypass an absent auxiliary tank station and go to the next occupied tank station.

A PUMP/VALVE FAIL caution light illuminates to show failure of any element(s) of the dual transfer/shutoff valves or dual transfer pumps. Normally, only a single valve and pump of the dual pump/valve system functions when fuel transfer is activated. Each valve and pump activates alternately with the other valve and pump to spread the use on the equipment. When a valve or pump fails and fuel flow is stopped, the PUMP/VALVE FAIL caution light will illuminate after approximately 40 seconds and the alternate valve will open. After an additional 40 seconds, if fuel flow has not begun, the alternate pump will start. After an additional 40 seconds, if fuel flow has still not started, the AUX FUEL XFER FAULT caution light illuminates to show total failure of the auxiliary fuel transfer system. The pressure sensor initiating the PUMP/VALVE FAIL light latches open and the light will not go out until the fault is repaired. An AUX FUEL XFER FAULT caution light illuminates when a transfer command in the FMCP for transfer to the main tank is received and auxiliary fuel transfer is not activated within approximately 120 seconds.

2.4.6 External Tank Jettison

The emergency panel marked EMER PNL (Figure 1-7) has a center switch marked ALL STORES SONO under the heading JETTISON. With weight-off-wheels, activating the switch will electrically fire all BRU-14 squib circuits, and all pylon external auxiliary bomb racks will release their stores.

2.5 AUXILIARY POWER UNIT SYSTEM

The APU system provides pneumatic power for starting the engines and operating the environmental control system (ECS). It incorporates a generator for ground and emergency in-flight electrical operations.

2.5.1 APU

The APU (Figure 2-19) is a gas turbine engine consisting of a power section, a reduction gearbox, appropriate controls, and accessories. The APU accessory gear box provides a mounting pad for the hydraulic starter and an output driver for the APU fuel assembly, oil pump, and air-cooled AC generator. The APU is lubricated by a self-contained oil system. Fuel consumption is 150 pounds per hour.

2.5.2 APU Accessories

APU system accessories include a prime/boost pump, hydraulic accumulator (with hand pump), hydraulic starter, and AC generator. The prime/boost pump is used to prime the engine or APU fuel lines and provides fuel under pressure to the APU during starting and operations at pressure altitudes at or above 8,000 feet. The hydraulic accumulator provides the hydraulic pressure for driving the APU starter. The minimum accumulator pressure required for starting the APU is approximately 2,650 psi. It can be recharged by using the accumulator hand pump. With AC power available, the accumulator is charged by the backup hydraulic pump.

2.5.3 APU Controls

The APU CONTR switch, located on the upper console. ON opens the APU airframe fuel shutoff valve and sends a start signal to the APU electronic sequence unit (ESU) or digital electronic sequence unit (DESU). OFF removes electrical power from the system closing the airframe fuel shutoff valve.

2.5.4 APU Control and Monitoring

The APU is controlled and monitored by the ESU/DESU. If a start sequence fails or a monitored parameter is exceeded during operation (with the exception of APU OIL TEMP HI), the ESU/DESU will automatically shut down the APU.

The ESU/DESU displays APU faults using built-in-test (BIT) indicators. The BIT indicators are capable of displaying start sequence or operation status and specific reasons for APU shutdown. To maintain BIT codes after failure, DC power is required and the APU CONTR switch must remain in the ON position. Four caution/advisories (APU ON, APU FAIL, APU OIL TEMP HI, and APU ACCUM LOW) provide monitoring of APU operation. APU FAIL indicates the APU has failed due to high Ng, low Ng, high TGT, low TGT, low oil pressure, or start sequence failure. APU OIL TEMP HI indicates the APU has reached the maximum oil temperature for continuous operation. The DESU for the Turbomach APU has the added capability to control APU overtemps by regulating the main fuel valve and start bypass valve.

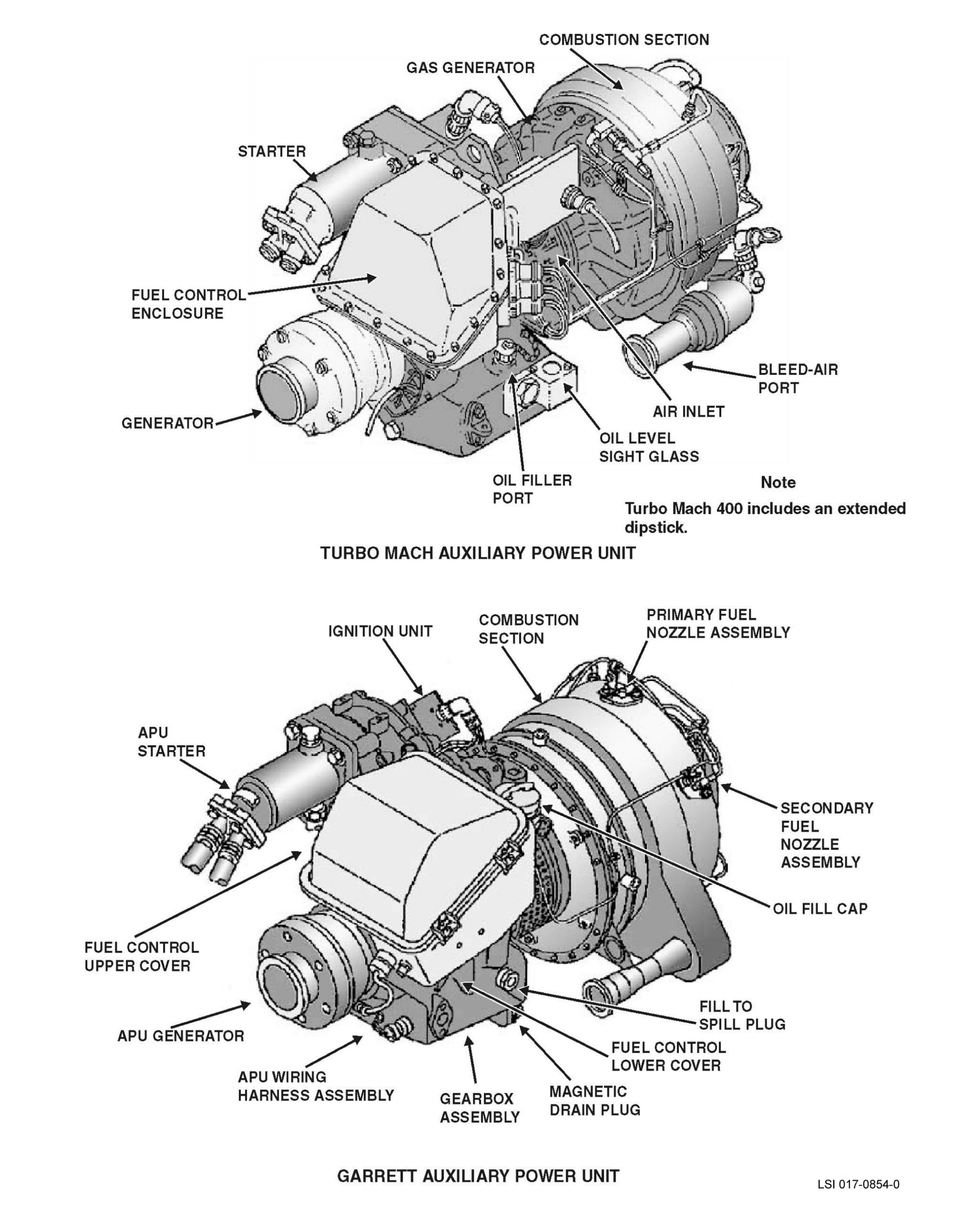


Figure 2-19. Auxiliary Power Units

2.5.5 APU Start System

With the FUEL PUMP switch in the APU BOOST position, pressurized fuel is supplied from the right fuel tank by the prime/boost pump. The fuel control governs and meters fuel flow to the APU power section, permitting automatic starting under all ambient conditions and constant speed operation once the APU has accelerated to its normal speed.

Placing the APU CONTR switch to ON initiates the start sequence. The ESU/DESU sends a signal to open the APU start valve, releasing the hydraulic accumulator charge to the starter. As the accumulator pressure drops below 2,650 psi, the APU ACCUM LOW advisory appears, indicating that the accumulator pressure is low. The APU ON advisory appears when the APU is on and operating normally. Placing the APU GENERATOR switch to ON makes electrical power available. If the backup pump is cycled ON then to the OFF or AUTO position, it will remain on for one cycle of 90 seconds (180 seconds with winterization kit installed). Once the accumulator is recharged, the APU ACCUM LOW advisory will extinguish.

If the APU does not start and the APU ACCUM LOW advisory is not illuminated, a start may be attempted by simultaneously moving the APU CONTR switch to ON and actuating the manual START/OVERRIDE lever located on the accumulator manifold. APU accumulator pressure will be dumped to the starter to turn the compressor until the APU has reached a self-sustaining speed.

2.6 ELECTRICAL SYSTEM

The primary source of electrical power for the SH-60B is alternating current (AC). There are three AC sources for the aircraft. The primary sources are the two transmission-driven main generators. The secondary source is the APU-driven generator. External AC power can also be connected to the helicopter. DC electrical power is obtained by two converters which convert AC power to DC power and reduce the voltage. A battery is installed for use in starting the APU and as a secondary source of DC power. AC and DC power are distributed to individual components by means of a bus distribution system.

2.6.1 AC Electrical System

The primary AC electrical power is supplied by two oil-cooled 30/45 kVA, 115 Vac, 3 phase, 400 Hz brushless generators, driven by the transmission through the accessory modules. The generators share their oil supply with the accessory modules, input modules, and main transmission. A secondary electrical power source is supplied by an air-cooled, 20/30 kVA, 115 Vac, 3 phase, 400 Hz brushless generator mounted on and driven by the APU. The generators are controlled by generator control units (GCUs). The GCUs regulate generator output and protect against overvoltage, undervoltage, underfrequency on the ground, and feeder fault for detection of open or short circuited feeder lines. In flight, the generators will remain online until N_r decreases to approximately 80 percent. A minimum of 97 percent N_r is required for the GCU to connect the NO. 1 and NO. 2 generators to the AC distribution system. The external power receptacle, which may be used to supply external AC power to the electrical system, is located on the right side of the aircraft, forward of the cabin door near the main mount. External power is monitored by the external power monitor panel located in the right-hand junction box. The external power source is monitored for phase rotation, overvoltage, undervoltage, underfrequency, and overfrequency to determine if the source is acceptable.

There are five buses in the AC electrical distribution system: the NO. 1 and NO. 2 AC primary buses (NO. 1/2 AC PRI BUS), the AC essential bus (AC ESNTL BUS), the AC secondary bus (AC SEC BUS), and the AC monitor bus (AC MON BUS). AC bus distribution loads are illustrated in (Figure 2-20). With both main generators operating, the NO. 1 generator powers the NO. 1 AC primary, AC essential, and AC secondary buses, while the NO. 2 AC generator powers the NO. 2 AC primary and the AC monitor bus. If the APU generator is selected while both main generators are operating, the APU generator will not be connected to the AC bus distribution system.

Should either main generator fail, automatic bus switching compensates by limiting the AC load to the available generator output. If combined current demand exceeds the capability of the operating generator(s), the buses are redistributed to available generators so that major bus loads can be managed as follows:

1. The backup hydraulic pump is the major load for the NO. 1 AC primary bus and has the highest priority.
2. The mission avionics system is the major load on the AC secondary bus and is the next priority. Tail rotor de-ice power is also supplied from this bus.

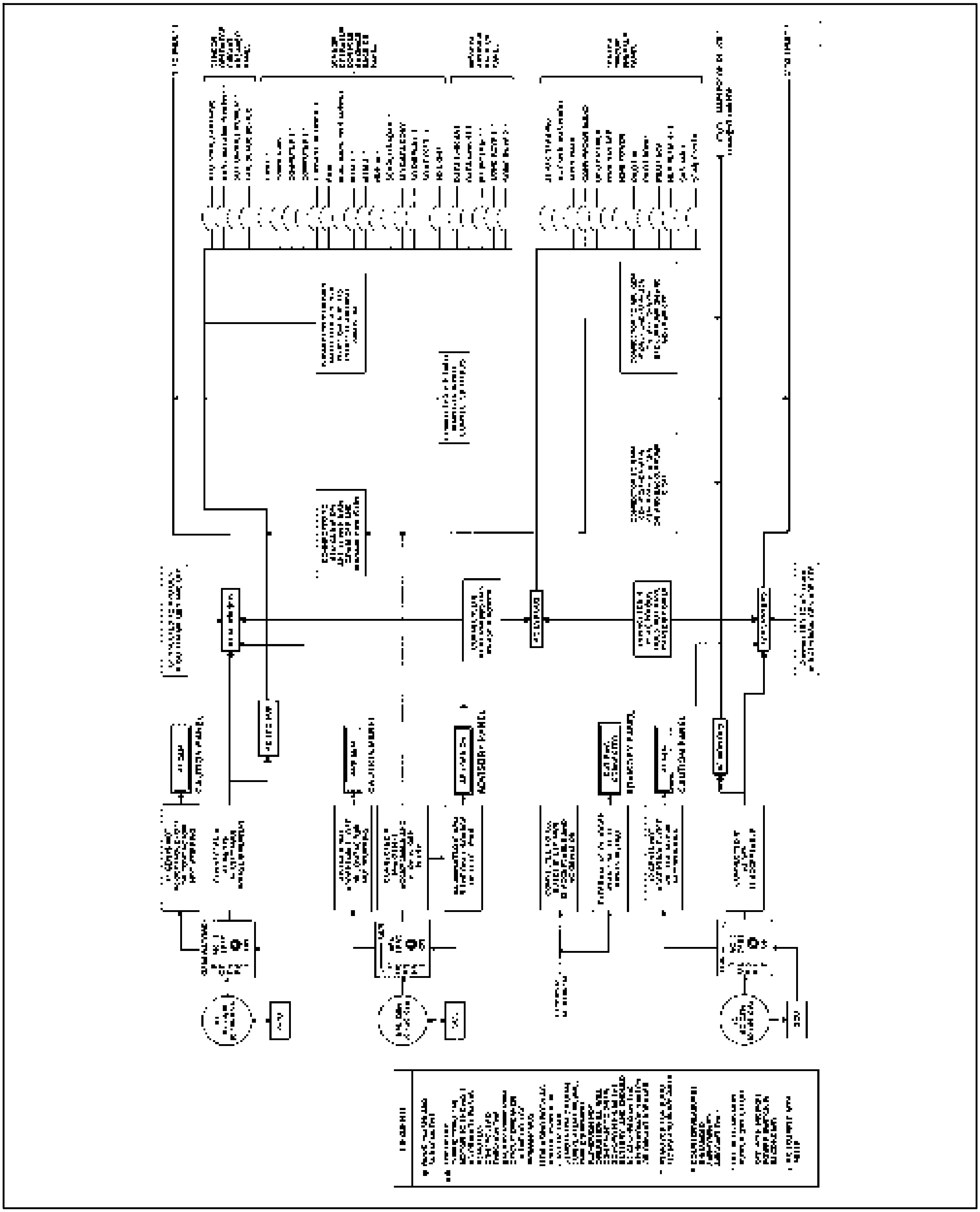


Figure 2-20. Electrical System Block Diagram (AC, Sheet 1 of 3)

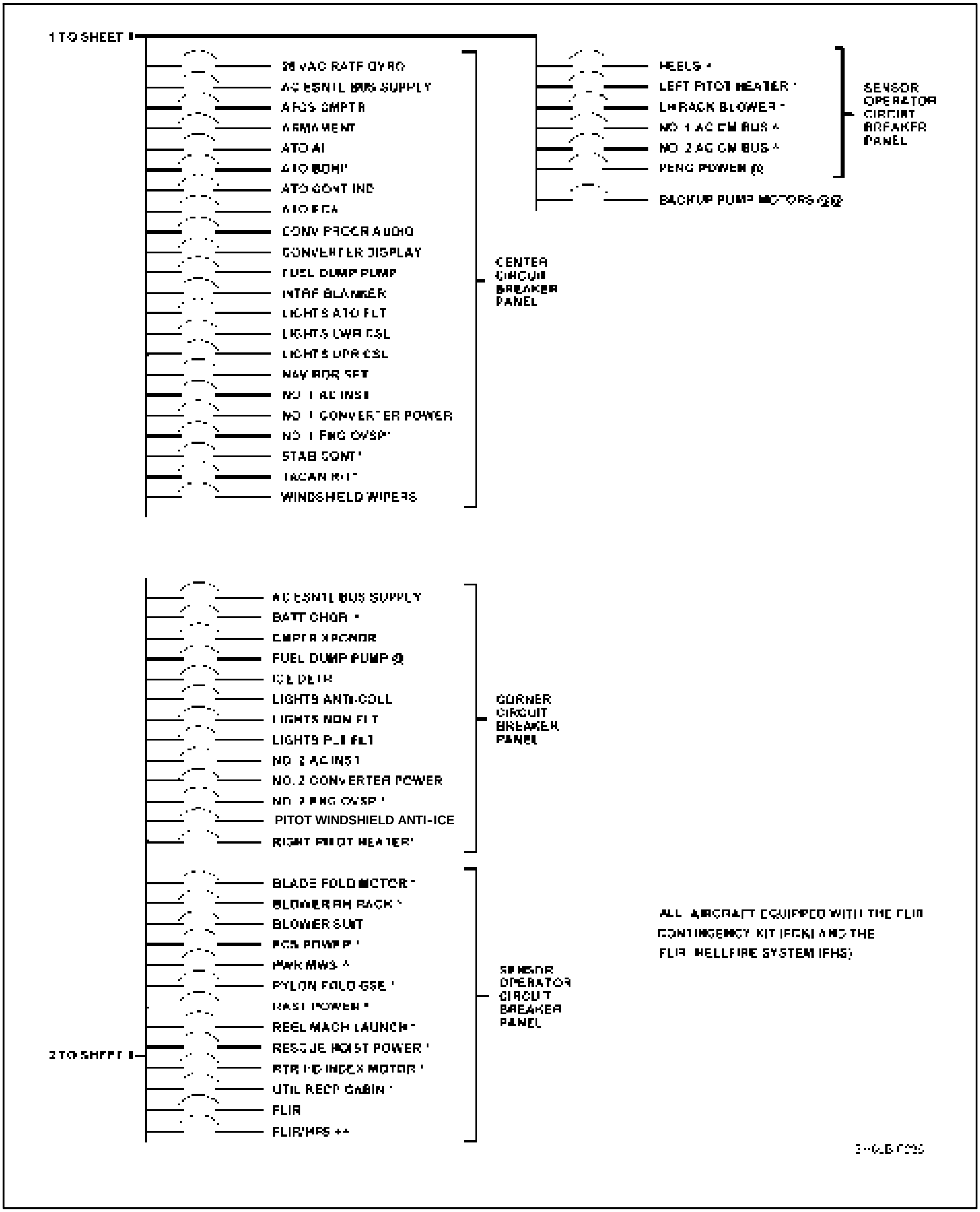


Figure 2-20. Electrical System Block Diagram (AC, Sheet 2)

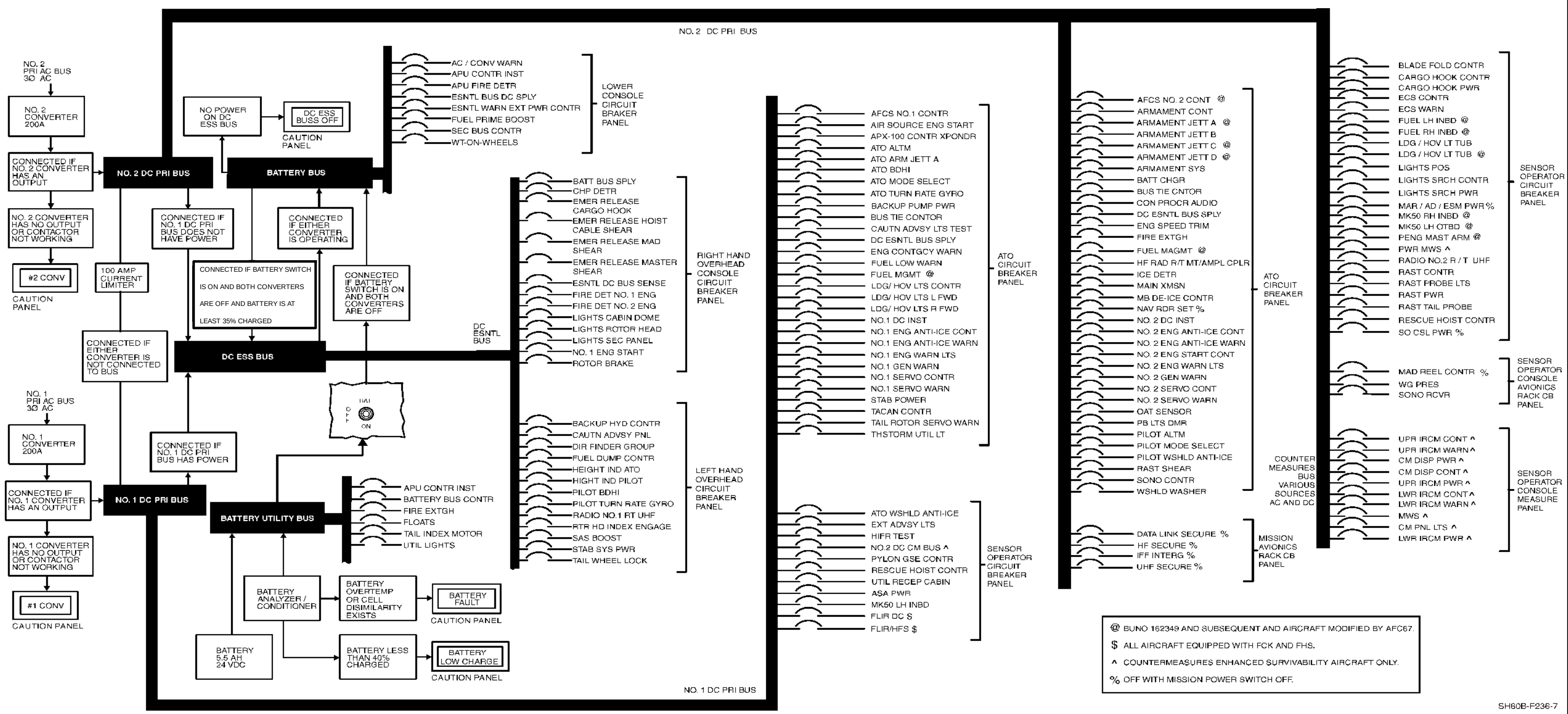


Figure 2-20. Electrical System Block Diagram (DC, Sheet 3)

3. The main rotor de-ice system is the only system powered from the AC monitor bus and has the lowest priority of the major current drawing components. AC bus distribution during normal and degraded modes is illustrated in Figure 2-21.

Cockpit switches for control of the generators and external power are located in the center overhead console. The GENERATORS, APU, NO. 1, and NO. 2 switches are three-position switches, labeled ON, RESET OFF, and TEST. The ON position energizes the generator and permits connection of the generator AC output to the distribution system. The RESET OFF position deenergizes the generator and permits generator recycling if the generator was disabled or disconnected from the distribution system. The TEST position permits testing of the AC output of the generator without connecting it to the distribution system. If generator output is normal, the generator caution light will not be illuminated. The EXT PWR switch is a three-position switch, labeled ON, OFF, and RESET. The ON position permits connection of external AC power to the distribution system. The OFF position disconnects external AC power from the distribution system. The RESET position permits recycling if the AC external source was unacceptable. External power will automatically be dropped from the aircraft distribution system when either main generator or the APU generator is brought on line. Mission systems will be lost and secure electrical keys may be lost with only the APU generator on line.

Illumination of the #1 GEN, #2 GEN, or APU GEN caution light indicates a failure of the respective generator, GCU, generator contactor, or a fault in the respective distribution system due to an overvoltage, undervoltage, underfrequency on the ground, or feeder fault. Illumination of the #1 or #2 GEN BRG caution light indicates a worn or failed main bearing on the respective generator.

WARNING

If APU is unavailable or external power is not accepted when main generators are secured, a total loss of AC power will occur. Systems lost include ICS, VIDS display, and AFCS computer power. When AFCS computer power is interrupted, trim is disengaged and an unguarded cyclic will allow the rotor arc to dip to as low as four feet above the deck.

Power to illuminate the generator caution lights is provided from the NO. 1 and NO. 2 DC primary buses through the NO. 1 GEN WARN and NO. 2 GEN WARN circuit breakers, respectively. Both circuit breakers are on the ATO circuit breaker panel. Illumination of the AC ESS BUS OFF caution light indicates a power loss on the AC essential bus. The caution light is powered by any source of DC power through a circuit breaker marked AC ESNTL BUS WARN and located on the center circuit breaker panel. If the APU is the only source of AC power, illumination of the APU GEN ON advisory light indicates that the APU-driven generator is on and supplying power to the system. Illumination of the EXT PWR CONNECTED advisory light indicates that the external power cable is connected to the helicopter and DC power is on the battery bus. The advisory light is powered by the battery bus through a circuit breaker on the lower console circuit breaker panel marked ESNTL WARN EXT PWR CONTR.

POWER SOURCE	#1 AC PRI	AC SEC	AC MONITOR
MAJOR LOAD ON BUS	BACKUP PUMP	MISSION POWER	BLADE DE-ICE
External Power	Available	Available	Available
Both Main Generators	Available	Available	Available
1 Main and APU Generator	Note 1	Note 1	Note 1
1 Main Generator	Note 2	Note 2	Not Available
APU Generator	Available	Not Available	Not Available

Notes:

1. Combination of any two.
2. Either system.

Figure 2-21. AC Bus Distribution

2.6.2 AC Bus Tie System

The AC bus tie contactor/relay (K4) connects APU power or external power to the primary AC buses. In addition, the K4 allows the output from the #1 generator to power the buses of a failed #2 generator. With both main generators online and supplying output through K1 (#1 Gen) and K2 (#2 Gen), K4 is deenergized. It is by design that when one generator fails, the operating generator will inherit the required load via a relay. This is accomplished in two ways:

1. In the case of # 1 generator failure, the #2 generator has a direct feed (not K4) straight to the K1 contactor allowing the #1 generator buses to continue operation despite the failure of the generator.

2. When the #2 generator fails, output from the #1 generator is routed through the AC bus tie (K4), thus keeping the #2 generator buses powered.

With the loss of the AC bus tie, no caution light nor change in cockpit functionality exists that would alert crews to identify the failure. Identification will become apparent when a second failure such as a generator or converter failure occurs. The following shows two scenarios:

1. With the failure of both the K4 contactor and the #2 AC generator, the #2 primary AC bus will be lost. All other bus ties will remain functional.

2. With the loss of both the K4 contactor and the #1 AC generator, all bus ties will remain functional.

Note

In both cases, powering of the APU will result in the pickup of the AC Monitor bus only.

With the AC bus tie failure, indicated by a failed generator and no load pickup, the perceived correction is to start the APU and disconnect the operating generator. The perception here is that the APU powers the AC primary buses through the K4 feeder. If the AC bus fails, then the APU will not connect to the primary buses. Therefore, disconnecting of an operating generator will only compound the emergency from single generator operations to a self-induced total AC power failure.

WARNING

In the event of an AC bus tie failure, starting of the APU and disconnection of the operating generator has no effect in correcting the malfunction. Further, should the operating generator be taken off-line, a complete AC power failure is imminent.

2.6.3 DC Electrical System

DC power is supplied by two converters each rated at 28 Vdc, 200 amps continuous power. NO. 1 and NO. 2 converters are powered by the NO. 1 AC PRI BUS and NO. 2 AC PRI BUS through the NO. 1 CONVERTER POWER and NO. 2 CONVERTER POWER circuit breakers respectively. The NO. 1 converter is located in the left-hand junction box and the NO. 2 converter is located in the right-hand junction box. A 24 Vdc, 5.5 amp hours battery located in the ATO seat well provides a secondary or emergency source of DC power. A battery analyzer/conditioner located in the ATO seat well monitors the battery system for fault conditions and provides a battery charging capability. The analyzer system monitors battery charge, internal temperature, and cell conditions and will automatically disconnect DC loads from the battery or the battery from the charging circuit, as appropriate. The system charges the battery whenever AC power is available and the battery switch is on.

There are five buses in the DC electrical distribution system (Figure 2-20): the NO. 1 and NO. 2 DC primary buses (NO. 1/2 DC PRI BUS), the DC essential bus (DC ESNTL BUS), the battery bus (BATT BUS), and the battery utility bus (BATT UTIL BUS). Automatic bus switching provides maximum flexibility should a converter fail. The NO. 1 converter is the source of power for the NO. 1 DC primary bus, the DC essential bus, and the battery bus. The NO. 2 converter powers the NO. 2 DC primary bus. The battery powers the battery utility bus. Failure of one of the converters results in its loads being picked up by the other converter. If both converters fail, the battery provides a source of power to the battery utility bus, the battery bus (if the battery switch is on), and the DC essential bus (if the battery switch is on and the battery is above a 35 percent charge) through the circuit breaker marked ESNTL BUS DC SPLY and located on the lower console circuit breaker panel. The NO. 1 and NO. 2 DC primary buses are dropped.

Battery power is controlled by a two–position BATT switch located on the center overhead console labeled BATT with positions ON and OFF. The ON position connects the DC power output of the battery utility bus to the battery bus and provides input power to the analyzer/conditioner. When the helicopter converters are operating and BATT switch is ON, the charging circuit of the analyzer/conditioner receives AC and DC power. DC charging power is supplied from the NO. 2 DC primary bus through the BATT CHGR circuit breaker on the ATO circuit breaker panel. AC power is supplied from the NO. 2 AC primary bus through the BATT CHGR circuit breaker on the corner circuit breaker panel.

Indicator lights on the caution/advisory panel permit cockpit monitoring of the DC electrical system. Illumination of the #1 CONV or #2 CONV caution light indicates a failure of the respective converter or DC bus contactor. Power to light the CONV caution lights is provided from the battery bus through a circuit breaker marked ESNTL BUS AC/CONV WARN and located on the lower console circuit breaker panel. Illumination of the DC ESS BUS OFF caution light indicates a power loss on the DC essential bus. Illumination of the BATT LOW CHARGE caution light indicates that the battery is below a 40 percent state of charge. DC essential bus power is required to light this light.

Illumination of the BATTERY FAULT caution light indicates that a battery overtemperature or cell dissimilarity condition exists. When a battery overtemperature or cell dissimilarity condition exists, the battery is disconnected from the charging circuit. When the battery drops below a 35 percent state of charge, the DC essential bus will be disconnected from the battery to allow sufficient charge for APU starting. The DC essential bus will still be powered if either converter is on or external power is connected. Power to illuminate the BATTERY FAULT caution light is provided by the battery bus through the ESNTL WARN PWR CONTR circuit breaker.

WARNING

With no other source of DC power for the DC ESNTL BUS and the battery below 35 percent charge, the BATT LOW CHARGE light will not be on, battery power may not be sufficient to fire the fire extinguisher cartridge–activated device (CAD), and the fire warning system will not be operative for the main engines.

2.6.4 DC Bus Tie System

DC bus tie contactor (K7) provides a connection between the NO. 1 and NO. 2 DC primary buses. If one converter fails, the path is closed from the primary bus of the operating converter to energize the solenoid of the DC bus tie contactor. The energized contactor connects the output of the operating converter to the primary bus of the failed converter.

Regardless of the bus tie, no caution light nor change in cockpit functionality exists that would alert crews to identify the failure. Identification happens when another electrical component fails, such as a generator or converter, where the electrical load required is not picked up by the operating generator or converter.

2.6.5 Circuit Breaker Panels

Nine circuit breaker panels are located in the cockpit and cabin area (Figure 2-22). Two upper console (overhead) circuit breaker panels contain circuit breakers protecting the DC essential bus. The lower console circuit breaker panel contains circuit breakers protecting the battery bus and the battery utility bus. The corner circuit breaker panel contains circuit breakers protecting the NO. 2 AC primary bus. The ATO circuit breaker panel contains circuit breakers protecting the NO. 1 and NO. 2 DC primary buses. The center circuit breaker panel contains circuit breakers protecting the NO. 1 AC primary bus and the AC essential bus. The SO circuit breaker panel contains circuit breakers protecting the NO. 1 AC primary bus, the NO. 2 AC primary bus, the AC secondary bus, the NO. 1 DC primary bus, and the NO. 2 DC primary bus. The SO console avionics rack and the mission avionics rack circuit breaker panels contain circuit breakers protecting the AC secondary bus and the NO. 2 DC primary bus. See Figure 2-23 for an alphabetical list of the circuit breakers.

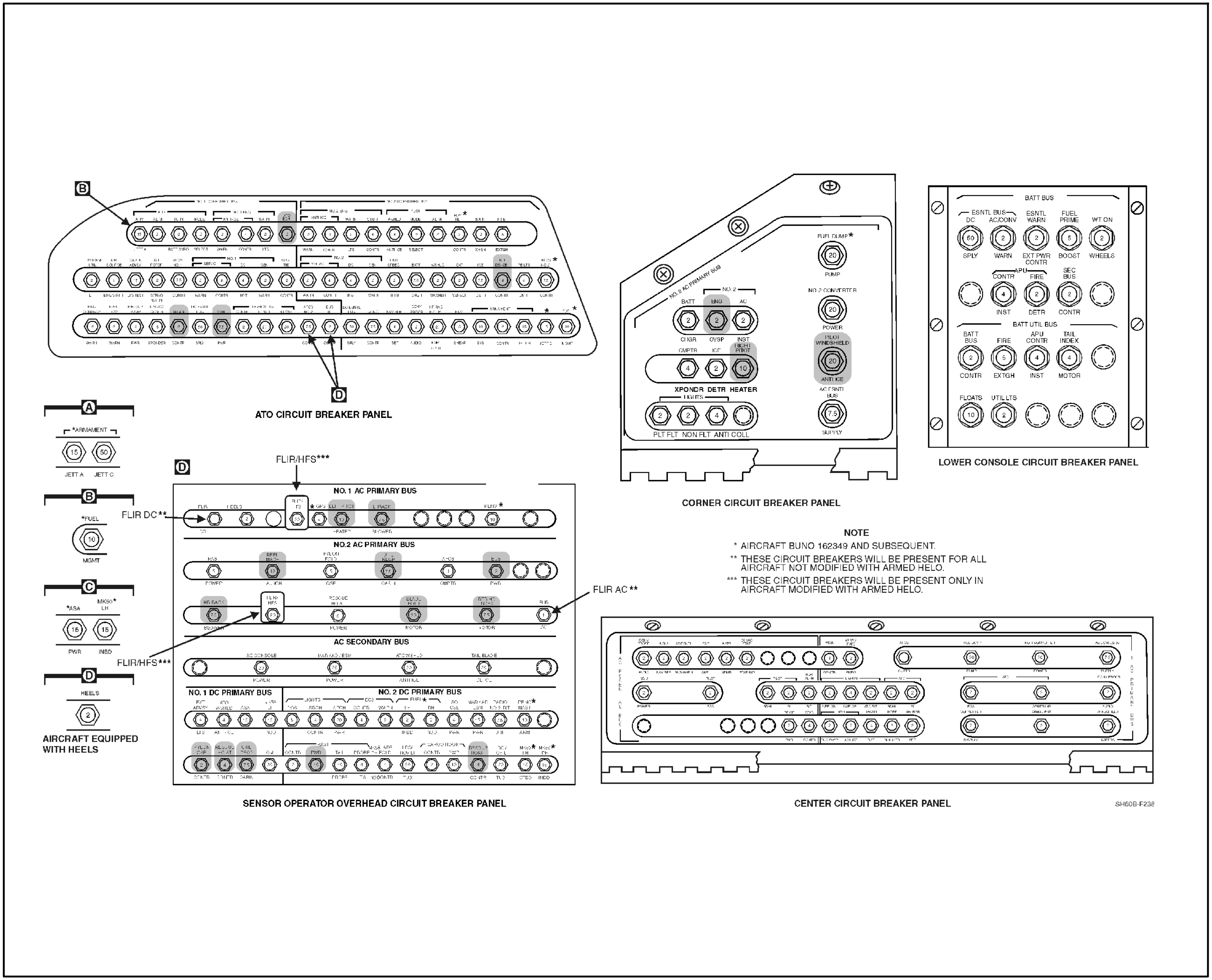


Figure 2-22. Circuit Breaker Panels (Sheet 1 of 2)

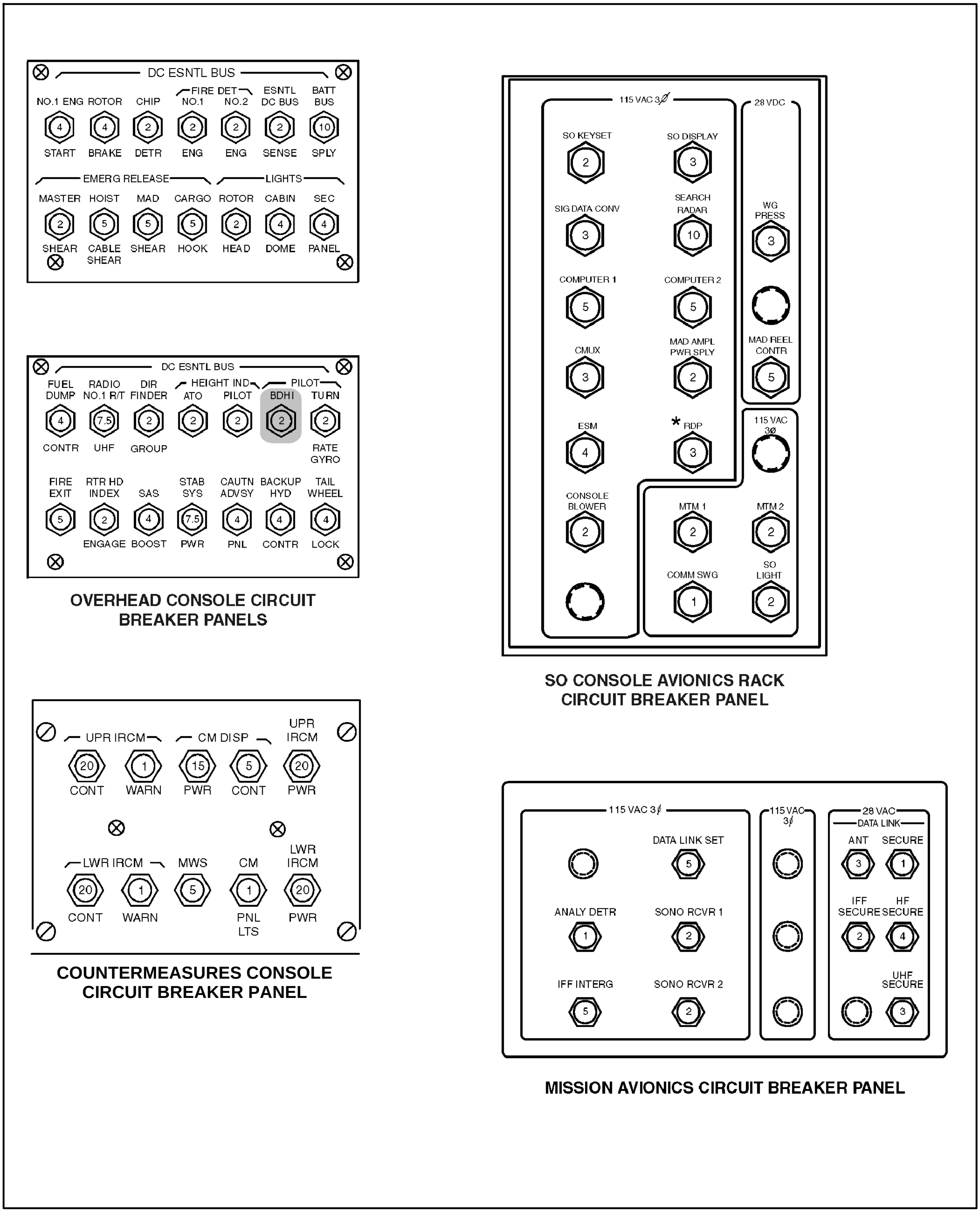


Figure 2-22. Circuit Breaker Panels (Sheet 2)

CIRCUIT BREAKER	BUS	LOCATION
AC ESNTL BUS SUPPLY	NO. 1 AC PRI BUS	CENTER
AC ESNTL BUS SUPPLY	NO. 2 AC PRI BUS	CORNER
AC ESNTL BUS WARN	AC ESS BUS	CENTER
AFCS CMPTR	NO. 2 AC PRI BUS	SO OVHD
AFCS CMTPR	NO. 1 AC PRI BUS	CENTER
AFCS NO. 1 CONTR	NO. 1 DC PRI BUS	ATO
AFCS NO. 2 CONTR	NO. 1 DC PRI BUS	ATO
AFCS NO. 2 CONTR	NO. 2 DC PRI BUS	ATO
AIR SOURCE ENG START	NO. 1 DC PRI BUS	ATO
ANALY DETR	AC SEC BUS	MISSION AVIONICS
APU CONTR INST	BATT UTIL BUS	LOWER CONSOLE
APU: CONTR INST	BATT BUS	LOWER CONSOLE
APU: FIRE DETR	BATT BUS	LOWER CONSOLE
APX–100 CONTR XPONDR	NO. 1 DC PRI BUS	ATO
ARMAMENT	NO. 1 AC PRI BUS	CENTER
ARMAMENT: CONTR	NO. 2 DC PRI BUS	ATO
ARMAMENT: JETT A	NO. 2 DC PRI BUS	ATO
ARMAMENT: JETT B	NO. 2 DC PRI BUS	ATO
ARMAMENT: JETT C	NO. 2 DC PRI BUS	ATO
ARMAMENT: SYS	NO. 2 DC PRI BUS	ATO
ASA	NO. 1 DC PRI BUS	SO OVHD
ASA PWR	NO. 1 DC PRI BUS	SO OVHD
ATO BDHI	NO. 1 DC PRI BUS	ATO
ATO WSHLD ANTI–ICE	NO. 1 DC PRI BUS	SO OVHD
ATO WSHLD ANTI–ICE	NO. 2 AC PRI BUS	SO OVHD
ATO:	NO. 1 AC PRI BUS	CENTER
ATO: ALTM	NO. 1 DC PRI BUS	ATO
ATO: ARM JETTA	NO. 1 DC PRI BUS	ATO
ATO: BDHI	NO. 1 AC PRI BUS	CENTER
ATO: CONTRIND	NO. 1 AC PRI BUS	CENTER
AUTO: EC	NO. 1 AC PRO BUS	CENTER
AUTO: MODE SELECT	NO. 1 DC PRO BUS	AUTO
AUTO: TURN RATE GYRO	NO. 1 DC PRO BUS	AUTO
AUTO FEMORA	AC ESS BUS	CENTER
BACKUP HYD CON TR	DC ESTEL BUS	OVHD CONSOLE
BACKUP PUMP PR	NO. 1 DC PRO BUS	AUTO
BATT BUS CON TR	BATT UT IL BUS	LOWER CONSOLE
BATT BUS SPLY	DC ESNTL BUS	OVHD CONSOLE
BATT CHGR	NO. 2 DC PRI BUS	ATO

Figure 2-23. Circuit Breaker List (Sheet 1 of 7)

CIRCUIT BREAKER	BUS	LOCATION
BATT CHGR	NO. 2 AC PRI BUS	CORNER
BLADE FOLD CONTR	NO. 2 DC PRI BUS	SO OVHD
BLADE FOLD MOTOR	NO. 2 AC PRI BUS	SO OVHD
BUS TIE CONTR	NO. 1 DC PRI BUS	ATO
BUS TIE CONTR	NO. 2 DC PRI BUS	ATO
CARGO HOOK: CONTR	NO. 2 DC PRI BUS	SO OVHD
CARGO HOOK: PWR	NO. 2 DC PRI BUS	SO OVHD
CAUTN ADVSY LTS TEST	NO. 1 DC PRI BUS	ATO
CAUTN ADVSY PNL	DC ESNTL BUS	OVHD CONSOLE
CHIP DETR	DC ESNTL BUS	OVHD CONSOLE
CM DISP: CONT	NO. 2 DC PRI BUS	SO COUNTERMEASURES
CM DISP: PWR	NO. 2 DC PRI BUS	SO COUNTERMEASURES
CM PNL LTS	NO. 2 DC PRI BUS	SO COUNTERMEASURES
CMPTR XPONDR	NO. 2 AC PRI BUS	CORNER
CMPTR XPONDR	NO. 1 DC PRI BUS	SO OVHD
CMUX	AC SEC BUS	SO CONSOLE AVIONICS RACK
COMM SWG	AC SEC BUS	SO CONSOLE AVIONICS RACK
COMPUTER 1	AC SEC BUS	SO CONSOLE AVIONICS RACK
COMPUTER 2	AC SEC BUS	SO CONSOLE AVIONICS RACK
CONSOLE BLOWER	AC SEC BUS	SO CONSOLE AVIONICS RACK
CONV PROCR AUDIO	NO. 2 DC PRI BUS	ATO
CONV PROCR AUDIO	AC ESS BUS	CENTER
CONV PROCR AUDIO	NO. 1 AC PRI BUS	CENTER
CONVERTER DISPLAY	NO. 1 AC PRI BUS	CENTER
DATA LINK SET	AC SEC BUS	MISSION AVIONICS
DATA LINK: ANT	NO. 2 DC PRI BUS	MISSION AVIONICS
DATA LINK: SECURE	NO. 2 DC PRI BUS	MISSION AVIONICS
DC ESNTL BUS SPLY	NO. 1 DC PRI BUS	ATO
DC ESNTL BUS SPLY	NO. 2 DC PRI BUS	ATO
DF GP PWR	AC ESS BUS	CENTER
DIR FINDER GROUP	DC ESNTL BUS	OVHD CONSOLE
ECS PWR	NO. 2 AC PRI BUS	SO OVHD
ECS: CONTR	NO. 2 DC PRI BUS	SO OVHD
ECS: WARN	NO. 2 DC PRI BUS	SO OVHD
EMERG RELEASE: CARGO HOOK	DC ESNTL BUS	OVHD CONSOLE
EMERG RELEASE: HOIST CABLE SHEAR	DC ESNTL BUS	OVHD CONSOLE
EMERG RELEASE: MAD SHEAR	DC ESNTL BUS	OVHD CONSOLE
EMERG RELEASE: MASTER SHEAR	DC ESNTL BUS	OVHD CONSOLE
ENG CONTGCY WARN	NO. 1 DC PRI BUS	ATO

Figure 2-23. Circuit Breaker List (Sheet 2)

CIRCUIT BREAKER	BUS	LOCATION
ENG SPEED TRIM	NO. 2 DC PRI BUS	ATO
ESM	AC SEC BUS	SO CONSOLE AVIONICS RACK
ESNTL BUS: AC/CONV WARN	BATT BUS	LOWER CONSOLE
ESNTL BUS: DC SPLY	BATT BUS	LOWER CONSOLE
ESNTL DC BUS SENSE	DC ESNTL BUS	OVHD CONSOLE
ESNTL WARN EXT PWR CONTR	BATT BUS	LOWER CONSOLE
EXT ADVSY LTS	NO. 1 DC PRI BUS	SO OVHD
FIRE DET: NO. 1 ENG	DC ESNTL BUS	OVHD CONSOLE
FIRE DET: NO. 2 ENG	DC ESNTL BUS	OVHD CONSOLE
FIRE EXIT	DC ESNTL BUS	OVHD CONSOLE
FIRE EXTGH	NO. 2 DC PRI BUS	ATO
FIRE EXTGH	BATT UTIL BUS	LOWER CONSOLE
FLIR AC	NO. 2 AC PRI BUS	SO OVHD
FLIR DC	NO. 1 AC PRI BUS	SO OVHD
FLIR/HFS	NO. 1 AC PRI BUS	SO OVHD
FLIR/HFS	NO. 2 AC PRI BUS	SO OVHD
FLOATS	BATT UTIL BUS	LOWER CONSOLE
FUEL DUMP CONTR	DC ESNTL BUS	OVHD CONSOLE
FUEL DUMP PUMP	NO. 1 AC PRI BUS	CENTER
FUEL DUMP PUMP	NO. 2 AC PRI BUS	CORNER
FUEL LOW WARN	NO. 1 DC PRI BUS	ATO
FUEL MGMT	NO. 2 DC PRI BUS	ATO
FUEL PRIME BOOST	BATT BUS	LOWER CONSOLE
FUEL: LH INBD	NO. 2 DC PRI BUS	SO OVHD
FUEL: RH INBD	NO. 2 DC PRI BUS	SO OVHD
GPS	NO. 1 AC PRI BUS	SO OVHD
HEELS	NO. 1 AC PRI BUS	SO OVHD
HEIGHT IND: ATO	DC ESNTL BUS	OVHD CONSOLE
HEIGHT IND: PILOT	DC ESNTL BUS	OVHD CONSOLE
HF RAD R/T MT AMPL CPLR	NO. 2 DC PRI BUS	ATO
HF SECURE	NO. 2 DC PRI BUS	MISSION AVIONICS
ICE DETR	NO. 2 DC PRI BUS	ATO
ICE DETR	NO. 2 AC PRI BUS	CORNER
IFF INTERG	AC SEC BUS	MISSION AVIONICS
IFF SECURE	NO. 2 DC PRI BUS	MISSION AVIONICS
INTRF BLANKER	NO. 1 AC PRI BUS	CENTER
JETT D	NO. 2 DC PRI BUS	ATO
LDG/HOV LT TUB	NO. 2 DC PRI BUS	SO OVHD
LDG/HOV LT TUB	NO. 2 DC PRI BUS	SO OVHD

Figure 2-23. Circuit Breaker List (Sheet 3)

CIRCUIT BREAKER	BUS	LOCATION
LDG/HOV LTS: CONTR	NO. 1 DC PRI BUS	ATO
LDG/HOV LTS: L FWD	NO. 1 DC PRI BUS	ATO
LDG/HOV LTS: R FWD	NO. 1 DC PRI BUS	ATO
LEFT PITOT HEATER	NO. 1 AC PRI BUS	SO OVHD
LH RACK BLOWER	NO. 1 AC PRI BUS	SO OVHD
LIGHTS: ANTI COLL	NO. 2 AC PRI BUS	CORNER
LIGHTS: ATO FLT	NO. 1 AC PRI BUS	CENTER
LIGHTS: CABIN DOME	DC ESNTL BUS	OVHD CONSOLE
LIGHTS: LWR CSL	NO. 1 AC PRI BUS	CENTER
LIGHTS: NON FLT	NO. 2 AC PRI BUS	CORNER
LIGHTS: PLT FLT	NO. 2 AC PRI BUS	CORNER
LIGHTS: POS	NO. 2 DC PRI BUS	SO OVHD
LIGHTS: ROTOR HEAD	DC ESNTL BUS	OVHD CONSOLE
LIGHTS: SEC PANEL	DC ESNTL BUS	OVHD CONSOLE
LIGHTS: SRCH CONTR	NO. 2 DC PRI BUS	SO OVHD
LIGHTS: SRCH PWR	NO. 2 DC PRI BUS	SO OVHD
LIGHTS: UPR CSL	NO. 1 AC PRI BUS	CENTER
LWR IRCM PWR	NO. 2 DC PRI BUS	SO COUNTERMEASURES
LWR IRCM: CONT	NO. 2 DC PRI BUS	SO COUNTERMEASURES
LWR IRCM: WARN	NO. 2 DC PRI BUS	SO COUNTERMEASURES
MAD AMPL PWR SPLY	AC SEC BUS	SO CONSOLE AVIONICS RACK
MAD REEL CONTR	NO. 2 DC PRI BUS	SO CONSOLE AVIONICS RACK
MAIN XMSN	NO. 2 DC PRI BUS	ATO
MAR/AD/ESM POWER	NO. 2 AC PRI BUS	SO OVHD
MAR/AD ESM PWR	NO. 2 DC PRI BUS	SO OVHD
MB DE–ICE CONTR	NO. 2 DC PRI BUS	ATO
MK50 LH INBD	NO. 1 DC PRI BUS	SO OVHD
MK50 LH OTBD	NO. 2 DC PRI BUS	SO OVHD
MK50 RH INBD	NO. 2 DC PRI BUS	SO OVHD
MK50 RH OTBD	NO. 2 DC PRI BUS	SO OVHD
MTM 1	AC SEC BUS	SO CONSOLE AVIONICS RACK
MTM 2	AC SEC BUS	SO CONSOLE AVIONICS RACK
MWS	NO. 2 DC PRI BUS	SO COUNTER MEASURES
NAV RDR SET	NO. 2 DC PRI BUS	ATO
NAV RDR SET	NO. 1 AC PRI BUS	CENTER
NO. 1 CONVERTER POWER	NO. 1 AC PRI BUS	CENTER
NO. 1 ENG START	DC ESNTL BUS	OVHD CONSOLE
NO. 1 ENG: ANTI–ICE: CONTR	NO. 1 DC PRI BUS	ATO
NO. 1 ENG: ANTI–ICE: WARN	NO. 1 DC PRI BUS	ATO

Figure 2-23. Circuit Breaker List (Sheet 4)

CIRCUIT BREAKER	BUS	LOCATION
NO. 1 ENG: WARN LTS	NO. 1 DC PRI BUS	ATO
NO. 1 AC INST	NO. 1 AC PRI BUS	CENTER
NO. 1 DC INST	NO. 1 DC PRI BUS	ATO
NO. 1 ENG OVSP	NO. 1 AC PRI BUS	CENTER
NO. 1 GEN WARN	NO. 1 DC PRI BUS	ATO
NO. 1 SERVO: CONTR	NO. 1 DC PRI BUS	ATO
NO. 1 SERVO: WARN	NO. 1 DC PRI BUS	ATO
NO. 2 CONVERTER POWER	NO. 2 AC PRI BUS	CORNER
NO. 2 ENG: ANTI-ICE: CONTR	NO. 2 DC PRI BUS	ATO
NO. 2 ENG: ANTI-ICE: WARN	NO. 2 DC PRI BUS	ATO
NO. 2 ENG: START CONTR	NO. 2 DC PRI BUS	ATO
NO. 2 ENG: WARN LTS	NO. 2 DC PRI BUS	ATO
NO. 2 AC INST	NO. 2 AC PRI BUS	CORNER
NO. 2 DC INST	NO. 2 DC PRI BUS	ATO
NO. 2 ENG OVSP	NO. 2 AC PRI BUS	CORNER
NO. 2 GEN WARN	NO. 2 DC PRI BUS	ATO
NO. 2 SERVO: CONTR	NO. 2 DC PRI BUS	ATO
NO. 2 SERVO: WARN	NO. 2 DC PRI BUS	ATO
NSIU NAV REF	AC ESS BUS	CENTER
NSIU POWER	AC ESS BUS	CENTER
OAT SENSOR	NO. 2 DC PRI BUS	ATO
P ENG	NO. 1 AC PRI BUS	SO OVHD
PB LTS CMR	NO. 2 DC PRI BUS	ATO
PILOT ECA	AC ESS BUS	CENTER
PILOT: AI	AC ESS BUS	CENTER
PILOT: ALTM	NO. 2 DC PRI BUS	ATO
PILOT: BDHI	AC ESS BUS	CENTER
PILOT: BDHI	DC ESNTL BUS	OVHD CONSOLE
PILOT: MODE SELECT	NO. 2 DC PRI BUS	ATO
PILOT: TURN RATE GYRO	DC ESNTL BUS	OVHD CONSOLE
PILOT: WSHLD ANTI-ICE	NO. 2 DC PRI BUS	ATO
PILOT WINDSHIELD ANTI ICE	NO. 2 AC PRI BUS	CORNER
PYLON FOLD GSE	NO. 2 AC PRI BUS	SO OVHD
PYLON GSE CONTR	NO. 1 DC PRI BUS	SO OVHD
R ENG MAST ARM	NO. 2 DC PRI BUS	SO OVHD
RADIO NO. 1 R/T UHF	DC ESNTL BUS	OVHD CONSOLE
RADIO NO. 2 R/T UHF	NO. 2 DC PRI BUS	SO OVHD
RAST POWER	NO. 2 AC PRI BUS	SO OVHD
RAST SHEAR	NO. 2 DC PRI BUS	ATO

Figure 2-23. Circuit Breaker List (Sheet 5)

CIRCUIT BREAKER	BUS	LOCATION
RAST: CONTR	NO. 2 DC PRI BUS	SO OVHD
RAST: PROBE LTS	NO. 2 DC PRI BUS	SO OVHD
RAST: PWR	NO. 2 DC PRI BUS	SO OVHD
RAST: TAIL PROBE	NO. 2 DC PRI BUS	SO OVHD
RATE GYRO 26 VAC	NO. 1 AC PRI BUS	CENTER
RDP	AC SEC BUS	SO CONSOLE AVIONICS RACK
RDR ALTM R/T	AC ESS BUS	CENTER
REEL MACH LAUNCH	NO. 2 AC PRI BUS	SO OVHD
RESCUE HOIST CONTR	NO. 1 DC PRI BUS	SO OVHD
RESCUE HOIST CONTR	NO. 2 DC PRI BUS	SO OVHD
RESCUE HOIST POWER	NO. 2 AC PRI BUS	SO OVHD
RH RACK BLOWER	NO. 2 AC PRI BUS	SO OVHD
RIGHT PITOT HEATER	NO. 2 AC PRI BUS	CORNER
ROTOR BRAKE	DC ESNTL BUS	OVHD CONSOLE
RTR HD INDEX ENGAGE	DC ESNTL BUS	OVHD CONSOLE
RTR HD INDEX MOTOR	NO. 2 AC PRI BUS	SO OVHD
SAS AMPL	AC ESS BUS	CENTER
SAS BOOST	DC ESNTL BUS	OVHD CONSOLE
SEARCH RADAR	AC SEC BUS	SO CONSOLE AVIONICS RACK
SEC BUS CONTR	BATT BUS	LOWER CONSOLE
SIG DATA CONV	AC SEC BUS	SO CONSOLE AVIONICS RACK
SO CONSOLE POWER	NO. 2 AC PRI BUS	SO OVHD
SO CSL PWR	NO. 2 DC PRI BUS	SO OVHD
SO DISPLAY	AC SEC BUS	SO CONSOLE AVIONICS RACK
SO KEYSET	AC SEC BUS	SO CONSOLE AVIONICS RACK
SO LIGHT	AC SEC BUS	SO CONSOLE AVIONICS RACK
SONO CONTR	NO. 2 DC PRI BUS	ATO
SONO RCVR 1	AC SEC BUS	MISSION AVIONICS
SONO RCVR 2	AC SEC BUS	MISSION AVIONICS
STAB CONTR	AC ESS BUS	CENTER
STAB CONTR	NO. 1 AC PRI BUS	CENTER
STAB IND 26 VAC	AC ESS BUS	CENTER
STAB PWR	NO. 1 DC PRI BUS	ATO
STAB SYS PWR	DC ESNTL BUS	OVHD CONSOLE
TACAN CONTR	NO. 1 DC PRI BUS	ATO
TACAN R/T	NO. 1 AC PRI BUS	CENTER
TAIL BLADE DE–ICE	NO. 2 AC PRI BUS	SO OVHD
TAIL INDEX MOTOR	BATT UTIL BUS	LOWER CONSOLE
TAIL ROTOR SERVO WARN	NO. 1 DC PRI BUS	ATO
TAIL WHEEL LOCK	DC ESNTL BUS	OVHD CONSOLE
THSTORM UTIL LT	NO. 1 DC PRI BUS	ATO
UHF SECURE	NO. 2 DC PRI BUS	MISSION AVIONICS

Figure 2-23. Circuit Breaker List (Sheet 6)

CIRCUIT BREAKER	BUS	LOCATION
UPR IRCM PWR	NO. 2 DC PRI BUS	SO COUNTERMEASURES
UPR IRCM: CONT	NO. 2 DC PRI BUS	SO COUNTERMEASURES
UPR IRCM: WARN	NO. 2 DC PRI BUS	SO COUNTERMEASURES
UTIL LTS	BATT UTIL BUS	LOWER CONSOLE
UTIL RECP CABIN	NO. 1 DC PRI BUS	SO OVHD
UTIL RECP CABIN	NO. 2 AC PRI BUS	SO OVHD
WG PRESS	NO. 2 DC PRI BUS	SO CONSOLE AVIONICS RACK
WINDSHIELD WIPERS	NO. 1 AC PRI BUS	CENTER
WSHLD WASHER	NO. 2 DC PRI BUS	ATO
WT ON WHEELS	BATT BUS	LOWER CONSOLE

Figure 2-23. Circuit Breaker List (Sheet 7)

2.6.6 Utility and Test Receptacles

The utility and test receptacle panel is located on the left-hand bulkhead below the SO window. The panel contains 28 Vdc and 115 Vac utility power receptacles and the test receptacles for the NO. 1, NO. 2, and APU generators.

2.7 LIGHTING

2.7.1 Exterior Lighting

2.7.1.1 Anticollision Lights

The anticollision light system contains four strobes in two separate units, one beneath the aft fuselage and one on top of the aft pylon section (Figure 1-3). The lights are controlled by two switches on the overhead console (Figure 1-6) labeled ANTI COLLISION LIGHTS UPPER, BOTH, LOWER and DAY, OFF, NIGHT. The system consists of a dual power supply and two interchangeable day/night anticollision lights. The dual supply system provides separate outputs for the aft fuselage light and the pylon-mounted light. Each anticollision light assembly contains two lamps, a red lens for night operation, and a clear lens for day operation. The desired strobe(s) is selected by placing the switch at UPPER, LOWER, or BOTH. To discontinue operation of the anticollision light(s), the DAY-NIGHT switch is placed to OFF. Power to operate the anticollision light system is provided from the NO. 2 AC primary bus through a circuit breaker marked LIGHTS, ANTI COLL and located on the corner circuit breaker panel.

2.7.1.2 Position Lights

Position lights (Figure 1-3) are outboard of the left and right landing gear support and on the trailing edge of the tail vertical fin. The lights are red on the left, green on the right, and white on the tail. Control of the position lights is through the overhead console panel (Figure 1-6) containing two switches, marked POSITION LIGHTS, DIM-OFF-BRIGHT, and STEADY-FLASH. When the intensity switch is placed to DIM or BRIGHT, all three lights go on at once. If the STEADY-FLASH switch is placed to FLASH, the three lights flash between 70 and 90 times per minute. The STEADY position causes the lights to remain on continuously. Power to operate the position lights is provided by NO. 2 DC primary bus through a circuit breaker marked LIGHTS POS and located on the SO circuit breaker panel.

On aircraft BuNo 162349 and subsequent, the left position light functions on the left outboard pylon when it is installed. When the left pylon and its associated wiring receptacle are removed, the lighting function is automatically transferred to the left landing gear support.

2.7.1.3 Searchlight

The searchlight (Figure 1-3) is mounted on the right bottom of the nose section and is controlled from either collective.

Except when the pilot SRCH LT switch is in the STOW position, the searchlight may be selected and operated from either collective regardless of opposite switch position. Selection of the STOW position by the pilot disconnects the ATO SRCH LT and four-way search light control switches. Should both the pilot and ATO attempt to slew the searchlight simultaneously, the light will freeze position until one of the four-way search light control switches is released. The 450 watt light can be moved forward through a 120° arc from the stow position. It can also be turned 360° in either a right or left direction on its axis. The light is operated by a switch labeled SRCH LT ON, OFF, and STOW. Directional control of the light is provided through the four-position searchlight control switch, labeled EXT (extend), RET (retract), L (left), and R (right). When the SRCH LT switch is placed ON, the lamp will go on, arming the control switch. Placing the control switch to EXT causes the light beam to move forward at a rate of approximately 12° per second. If SRCH LT switch is placed to OFF, the light will extinguish and remain in its present position. If the switch is held at STOW, the light will retract at a rate of approximately 30° per second to the stowed position in the searchlight well. When the light is fully retracted, power is automatically removed. Power to light and control the searchlight is provided from the NO. 2 DC primary bus through circuit breakers, marked LIGHTS SRCH PWR and SRCH CONTR, located on the SO circuit breaker panel.

2.7.1.4 Landing/Hover Lights

Two 450 watt fixed-position landing/hover lights (Figure 1-3) are installed on the left and right sides beneath the nose section, and a fixed-position hover/rescue light is installed on the right side lower tub below the hoist. All lights are primarily controlled from the cockpit through the toggle switch, marked HOVER LIGHTS, ALL, OFF, and FWD (Figure 1-6). In the ALL position, nose hover lights and rescue light are turned on. At FWD, only the nose lights are turned on. The rescue light may be operated from the crew hover panel through the toggle switch, marked RESCUE LIGHT, ON, and OFF. The rescue light can be operated from the crew hover panel only when the cockpit switch is selected to OFF or FWD. Power for the forward hover lights is from the NO. 1 DC primary bus through circuit breakers marked LDG/HOV LTS CONTR, L FWD, and R FWD and located on the ATO circuit breaker panel. The rescue hoist light receives power from the NO. 1 DC primary bus through a circuit breaker marked LDG/HOV LT, CONTR on the ATO circuit breaker panel, and the NO. 2 DC primary bus through a circuit breaker marked LDG/HOV LT TUB on the SO circuit breaker panel.

2.7.1.5 Rotor Head Light

A light on top of the main transmission cabin fairing illuminates the rotor head droop stops (Figure 1-3). The rotor head light is controlled by a switch marked RTR HD LIGHTS, OFF, and ON (Figure 1-6). The light allows the director to determine the position of the droop stops during rotor engagement and disengagement. Power for the light is from the DC essential bus through a circuit breaker marked LIGHTS ROTOR HEAD and located on the overhead console circuit breaker panel.

2.7.1.6 Recovery Assist Secure and Traverse Lights

Recovery Assist Secure and Traverse (RAST) lights are located forward of the RAST probe. The lights are controlled by a single two-position switch marked RAST LIGHTS, ON, and OFF and located on the overhead console (Figure 1-6). Power to operate the RAST lights is provided by the NO. 2 DC primary bus through a circuit breaker located on the SO circuit breaker panel, under the general heading RAST, marked PROBE LTS.

2.7.2 Interior Lighting

The interior lighting system consists of the flight instrument and console lights, secondary lights, thunderstorm utility lights, utility lights, and cabin dome lights. Both AC and DC sources of electrical power are used to operate the various interior lights.

2.7.2.1 Flight Instrument and Console Lights

The flight instrument and console lights are the primary means for illuminating cockpit gauges and control indicators. These lights consist of individual gauge lights and backlit instrument panels. Instrument lights are grouped into flight and nonflight instruments. The flight instrument lights are divided into pilot and ATO. These lights are controlled by individual rotary intensity controls (Figure 1-6) marked INST LIGHT PILOT FLIGHT, OFF-BRIGHT, and INST

LIGHTS ATO FLIGHT, OFF–BRIGHT. The dimming control for the pilot flight instrument lights switches the caution/advisory panel lights from a bright to dim intensity when the switch is rotated out of the OFF position.

Power for the ATO instrument lights is supplied by the NO. 1 AC primary bus through the LIGHTS ATO FLT circuit breaker on the center circuit breaker panel. The NO. 2 AC primary bus powers the pilot flight instrument lights and the nonflight instrument lights through circuit breakers marked LIGHTS, PLT FLT, and NON FLT, respectively. Both circuit breakers are on the corner circuit breaker panel.

The nonflight and console lights operate in the same manner as the flight instrument lights. Intensity of the nonflight instrument lights is controlled by a rotary control, marked INST LIGHT NON FLIGHT, OFF–BRIGHT. Illumination of the upper and lower consoles is controlled by two rotary switches marked CONSOLE PANEL LTS UPPER, OFF–BRIGHT, and LOWER, OFF–BRIGHT. Power to operate the console lights is provided by the NO. 1 AC primary bus through two circuit breakers marked LIGHTS UPR CSL and LIGHTS LWR CSL, both on the center circuit breaker panel. Illumination intensity of the backlit pushbutton switches on the AFCS CONTROL panel and the AI/BDHI mode select control panels (pilot and ATO) is controlled by the CONSOLE PANEL LTS LOWER, OFF–BRIGHT rotary knob located on the ATO side of the upper console. Power for the lower console light switches is provided from the NO. 2 DC primary bus through the PB LTS DMR circuit breaker on the ATO circuit breaker panel.

2.7.2.2 Secondary Lights

The secondary lights system consists of DC powered floodlights that augment the AC powered flight instrument and console lights. Secondary lights also provide light for gauges and the lower console in the event of a total loss of AC power. Secondary illumination of the instrument panel is provided by five light fixtures mounted below the glare shield. Each fixture has a mechanical shade that allows the pilot/ATO to manually dim the light once the light has been turned on. A rotary knob on the ATO side of the overhead console marked INST PANEL SECONDARY LTS, OFF–BRIGHT controls the reference brightness of all five light fixtures. A white floodlight located above and behind the pilot seat provides a secondary means to illuminate the lower console. This light is controlled by a switch marked LOWER CONSOLE SECONDARY LT, DIM–OFF or BRT. Power for the secondary lights system is from the DC essential bus through a circuit breaker marked LIGHTS SEC PANEL and located on the overhead circuit breaker panel.

2.7.2.3 Thunderstorm Utility Lights

The thunderstorm utility lights help prevent pilot disorientation during night flight in thunderstorm conditions. When set to the BRIGHT position, the THUNDRSTRM/UTILITY LT switch turns on the thunderstorm utility light to its full intensity and overrides the dimming controls, bringing all other cockpit lights to full intensity. In the DIM position, the thunderstorm utility light and the other cockpit lighting are set to half intensity. Power to operate the thunderstorm utility lights is provided by the NO. 1 DC primary bus through a circuit breaker marked THSTORM UTIL LT and located on the ATO circuit breaker panel.

2.7.2.4 Utility Lights

Portable utility lights with coiled cords are attached to the upper cockpit bulkhead behind the pilot and ATO, and to the upper cabin bulkhead beside the SO, by removable brackets. The lights may be adjusted on their mountings to direct the light beams or they may be removed and used portably. The utility lights are controlled by a rheostat or a pushbutton on the end of each casing. The lens casing of the light may be turned to change from white light to red and spot to flood. The utility lights operate from the battery utility bus through a circuit breaker marked UTIL LTS on the lower console circuit breaker panel.

Note

Ensure utility lights are off when not in use to preclude unnecessary battery drain.

2.7.2.5 Cabin Dome Lights

Three cabin dome lights for lighting the cabin section are controlled by a CAB DOME LIGHT panel switch. On aircraft BuNo 162092 and subsequent, the panel switch has a RED position which turns the forward aisle light to red. The intensity controls are in the cabin on the crew hover panel and overhead of the SO. The controls are marked DOME LIGHT, OFF–BRT. The light level control may be adjusted to any position between the two extremes. Power to operate the cabin dome light system is provided from the DC essential bus through a circuit breaker marked LIGHTS CABIN DOME and located on the overhead circuit breaker panel.

2.8 HYDRAULIC SYSTEMS

The three hydraulic systems (Figure 2-25) are designed to provide full–flight control pressure (3,000 to 3,100 psi). The components of the hydraulic systems are three hydraulic pump modules, two transfer modules, a utility module, three dual–stage primary servos, one dual–stage tail rotor servo, three pilot–assist (boost) servos, four pilot–assist (SAS) servos, two hydromechanical trim actuators, the rescue hoist, an APU accumulator, an APU hand pump, and a servicing hand pump (Figure 2-26). There are three hydraulic pressure supply systems: NO. 1, NO. 2, and backup. All are completely independent, and each is fully capable of providing essential flight control pressure (3,000 to 3,100 psi) for maximum system redundancy. Complete redundancy is accomplished by the backup pump, providing hydraulic power to both NO. 1 and/or NO. 2 systems if one or both pumps fail. If NO. 1 and NO. 2 systems lose pressure, there will be a slight restriction in the maximum rate of flight control movement due to the backup pump supplying both primary stages with hydraulic power. When the SERVO switch, located on the pilot/ATO collective grips, is moved to the 1st OFF or 2nd OFF position, that stage of the primary servos is turned off. A malfunction in the other stage will cause the stage that was turned off to automatically come back on, provided the backup pump does not take over the functions of the lost system. A hydraulic hand pump is provided for APU accumulator pressurization in the event the backup pump is unavailable.

Note

The caution lights shown below may flicker when the listed switch is activated (Figure 2-24).

SUBSYSTEM	CAUTION
SAS 1 or SAS 2 switch on	#2 PRI SERVO PRESS #2 HYD PUMP BOOST SERVO OFF
SAS/BOOST HYD switch on	#2 PRI SERVO PRESS #2 HYD PUMP SAS
TAIL SERVO switch BKUP	#1 PRI SERVO PRESS #1 HYD PUMP
HYD LEAK TEST switch NORM after RESET	#1 and #2 PRI SERVO PRESS #1 and #2 HYD PUMP

Figure 2-24. Hydraulic System Activation

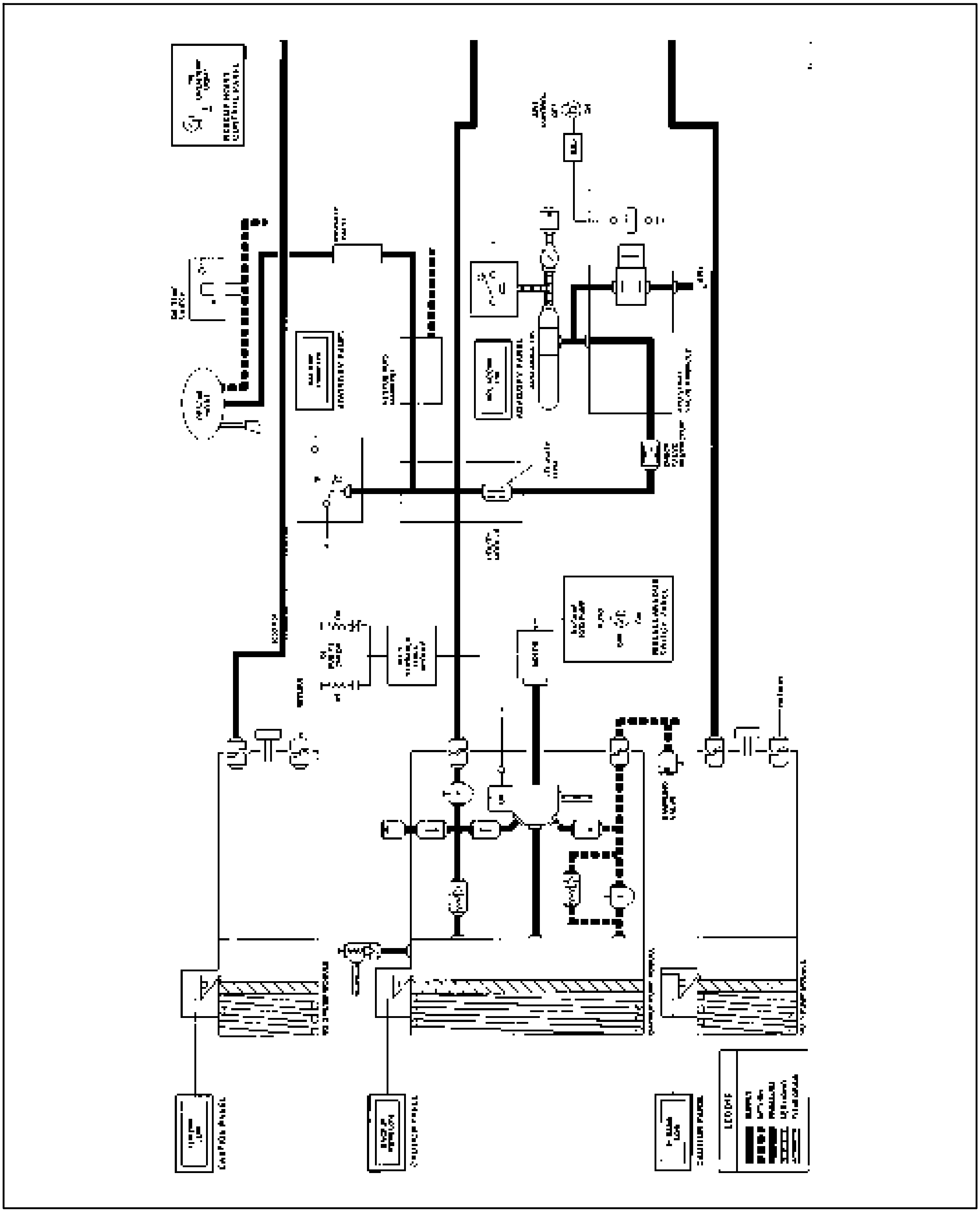


Figure 2-25. Flight Control Hydraulic System (Sheet 1 of 3)

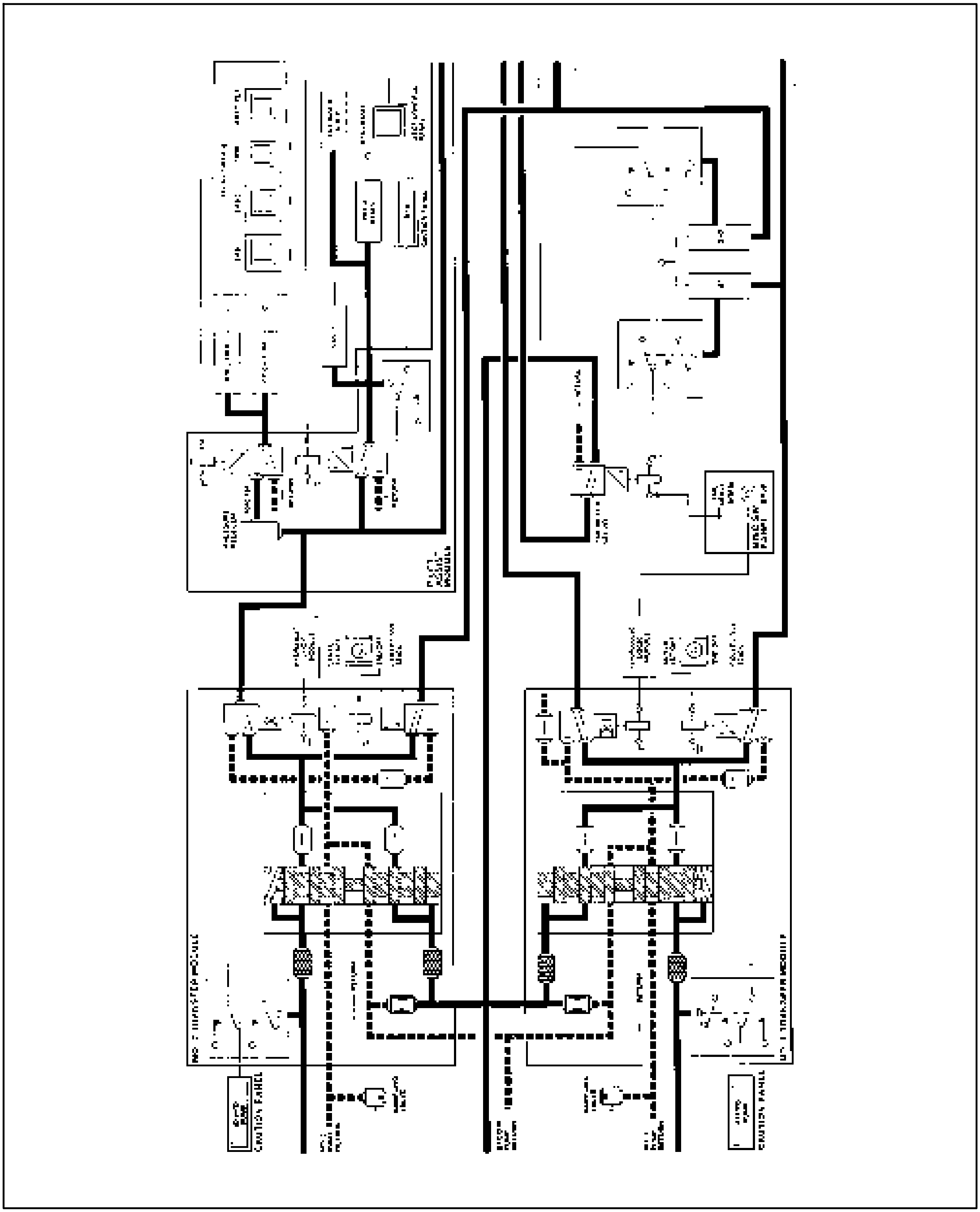


Figure 2-25. Flight Control Hydraulic System (Sheet 2)

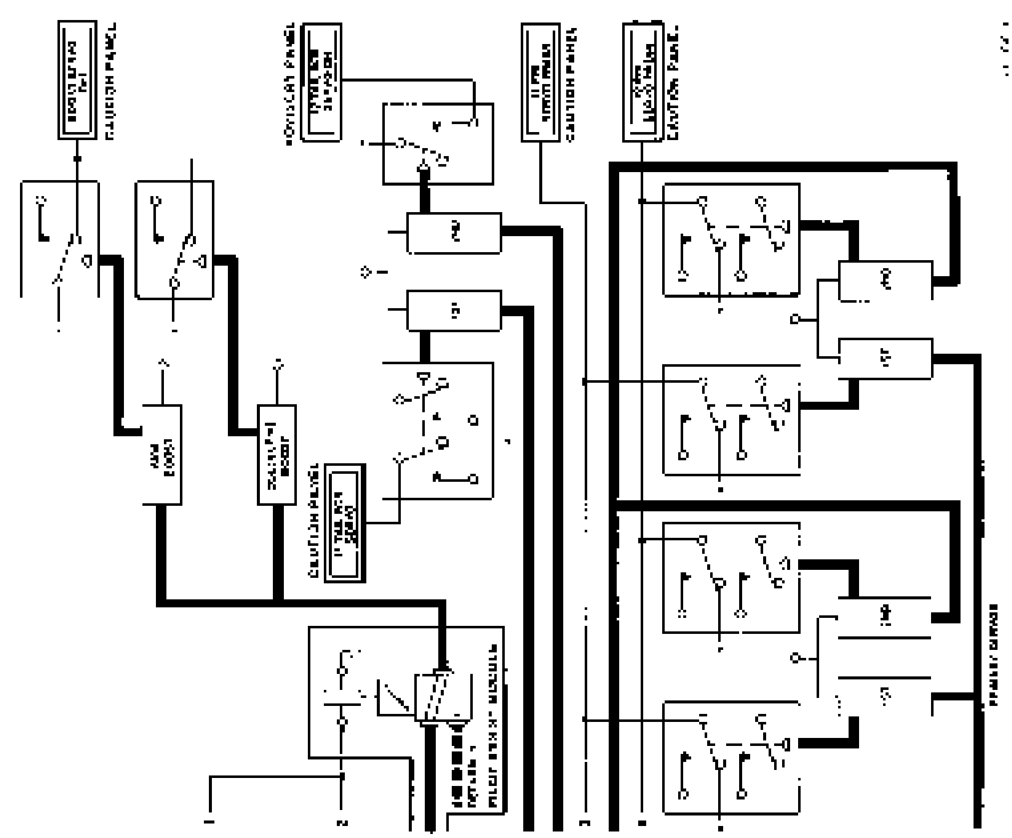


Figure 2-25. Flight Control Hydraulic System (Sheet 3)

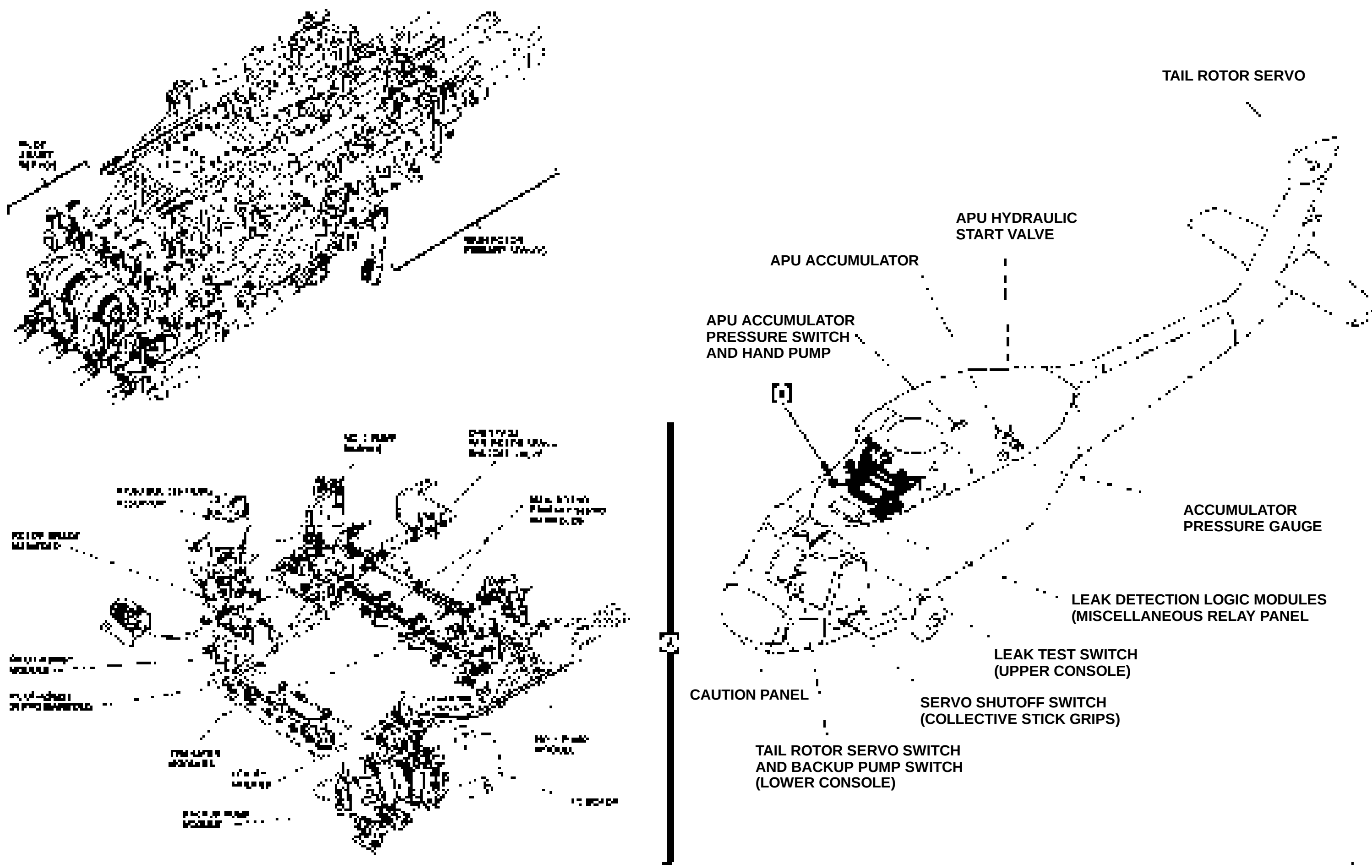


Figure 2-26. Hydraulics Bay

2.8.1 Hydraulic Pump Modules

The hydraulic pump modules are combination hydraulic pumps and reservoirs. The NO. 1, NO. 2, and backup pump modules are identical and interchangeable. The NO. 1 pump module is mounted on and driven by the left-accessory module of the main transmission. The NO. 2 pump module is mounted on and driven by the right-accessory transmission module. The backup pump module is mounted on and driven by an AC electric motor, powered by the NO. 1 AC primary bus. The reservoir part of each pump module has a level indicator. Markings correspond to underserviced, normal, and overserviced fluid levels. A pressure relief and bleed valve protects the pump from high pressure in the return system. Each pump has two filters: a pressure filter and a return filter. A red indicator button on each filter will pop out when pressure is 70 ±10 psi for the pressure filter, and 100 ±10 psi for the return filter. The pressure filter has no bypass. The return filter has a bypass valve that opens when return pressure reaches 100 ±10 psi. Each pump has three check valves: one at the external ground coupling, one at the pressure side, and one at the return side. A low level fluid indicator switch, mounted on top of each pump module, senses fluid loss for that system. When the piston on the pump module reaches the REFILL mark, the piston closes the switch, turning on a caution light marked RSVR LOW.

2.8.1.1 NO. 1 Hydraulic System

The system operates with the rotor turning and supplies the first stage of all primary servos and the first stage of the tail rotor servo. The system components are an integrated pump module, a transfer module, first-stage primary servos and first-stage tail rotor servo. The primary servos are controlled by the SERVO switch, located on the pilot/ATO collective grips. The switch can turn off either first or second stage of the primary servos, but not both at the same time. First-stage tail rotor servo can be manually turned off by a two-position switch, marked TAIL SERVO, NORM and BKUP on the miscellaneous switch panel (Figure 1-7).

2.8.1.2 NO. 2 Hydraulic System

The NO. 2 hydraulic system, which also operates with the rotor turning, supplies the second-stage primary servos and the pilot-assist servos. System components are the integrated pump module, transfer module, second-stage primary servos, and pilot-assist modules. Second-stage primary servos can be manually turned off by the SERVO switch. The pilot-assist servos cannot be turned off collectively, but SAS, TRIM, and BOOST servos can be manually turned off by switches on the AFCS CONTROL panel.

2.8.1.3 Backup Hydraulic System

This system supplies emergency pressure to the NO. 1 and/or NO. 2 hydraulic systems whenever a pressure loss occurs. It also supplies pressure to the NO. 2 stage of the tail rotor servo in case of a loss of pressure in the first stage of the tail rotor servo or #1 RSVR LOW indication. This system supplies utility hydraulic pressure to all flight control components during ground checkout. The backup system also provides 3,000 to 3,100 psi hydraulic pressure for recharging of the APU start system accumulator and for rescue hoist operation. The backup hydraulic system pump module is driven by an electric motor, which can be powered by any adequate three-phase AC power source. An internal depressurizing valve in the backup pump module reduces the output pressure of the pump to about 700 psi to aid startup of the electric motor. This valve unloads the electric motor by reducing the torque requirement at low rpm. After up to 4 seconds on APU or external power or 0.5 seconds with either main generator on, the valve is closed and 3,000 to 3,100 psi pressure is supplied to the hydraulic system. This sequence reduces the current demand during backup system startup.

CAUTION

If the BACKUP PUMP PWR circuit breaker is out and a condition exists which requires the backup pump to operate, then either the hydraulic system must be configured so that the backup pump will not activate upon resetting the circuit breaker, or AC power must be secured prior to resetting the circuit breaker. Damage to the current limiters may occur and will be indicated by a loss of all loads on NO. 1 AC primary bus.

Pressure–sensing switches in the NO. 1 and NO. 2 transfer modules constantly monitor the pressure output of the NO. 1 and NO. 2 pumps. Loss of pressure initiates the backup operation. The backup pump automatically operates when the BACKUP HYD PMP switch is in the AUTO position (or in any position while airborne) if any of the following criteria occurs:

1. NO. 1 hydraulic reservoir low.
2. NO. 1 tail rotor servo inoperative.
3. NO. 1 hydraulic pump failure.
4. NO. 2 hydraulic pump failure.

The system then provides emergency pressure to maintain full–flight control capability. A WOW switch on the left main landing gear provides automatic operation of the backup pump when the helicopter is in the air, regardless of BACKUP HYD PMP switch position. A pressure–sensing switch at the tail rotor monitors supply pressure to the first–stage tail rotor servo. The backup pump can supply pressure to either the first–stage or second–stage tail rotor servo if the NO. 1 pump loses pressure. This gives the pilot a backup tail rotor servo even with the loss of the primary hydraulic supply or #1 RSVR LOW. If a leak in a primary servo system depletes the backup system fluid, the backup reservoir–low level fluid indicator switch will turn on the BACK–UP RSVR LOW caution light.

The backup pump has a three–position toggle switch, located on the lower console (Figure 1-7). In the OFF position, the backup pump is not activated. In the AUTO position, the backup pump automatically maintains pressure in the NO. 1 or NO. 2 hydraulic systems as required and to the second–stage tail rotor servo. In the ON position with the NO. 1 and NO. 2 hydraulic modules operating normally, the backup pump recirculates hydraulic fluid to and from each transfer module and will maintain pressure in the APU accumulator. With the rotors turning and the APU accumulator low, once the backup pump is turned on it cannot be turned off until the accumulator is charged. With both hydraulic systems operating normally, once turned on, with the APU accumulator low, the backup pump will run for at least 90 seconds regardless of switch position (180 seconds with winterization kit installed). When airborne, with the BACKUP HYD PMP switch in any position, the backup pump maintains pressure in the NO. 1 or NO. 2 hydraulic systems and to the NO. 2 stage of the tail rotor servo as required. In the ON position in flight, the pump will remain on until secured.

CAUTION

Backup pump initiation with AFCS heading hold engaged (AFCS CONTROL TRIM switch ON, pedal microswitch NOT depressed) may cause AFCS Heading Hold failure, indicated by a compass fail flag, AFCS CONTROL panel HDG FAIL advisory, and a flashing AFCS DEGRADED caution light. Should this malfunction occur, system operation can be restored by pressing one of the fail advisory mode reset switches.

2.8.2 Transfer Modules

The NO. 1 and NO. 2 transfer modules connect hydraulic pressure from the pump modules to the flight control servos. Each interchangeable module is an integrated assembly of shutoff valves, pressure switches, check valves, and restrictor.

2.8.2.1 NO. 1 Transfer Module

This module has a transfer valve, a pressure switch, a first–stage primary shutoff valve, a first–stage tail rotor shutoff valve, a restrictor, and check valves. The transfer valve is spring loaded to the open or normal position. If the NO. 1 hydraulic system pressure is lost, the valve automatically transfers backup pump pressure to the NO. 1 system. The first–stage primary shutoff valve lets the pilot or ATO shut off first–stage pressure to the primary servos and prevents both stages from being shut off at the same time. The pressure switch lights the #1 HYD PUMP caution light on the caution/advisory panel when pressure drops to 2,000 psi and also sends a signal to a logic module that pressure is lost in the NO. 1 hydraulic system. The restrictor allows fluid to circulate for cooling under no-flow conditions. If a fluid leak develops past the transfer module, the check valves prevent fluid loss on the return side of the transfer module.

2.8.2.2 NO. 2 Transfer Module

This transfer module is like the NO. 1 module, except that it supplies NO. 2 hydraulic system pressure. The pilot-assist shutoff valve turns off pressure to the pilot-assist module. The second-stage primary servo shutoff valve turns off pressure to the second stage of the primary servos. The pressure switch turns on the #2 HYD PUMP caution light on the caution/advisory panel when the system pressure is below 2,000 psi and also sends a signal to a logic module that pressure is lost in the NO. 2 hydraulic system.

2.8.2.3 Utility Module

The utility module connects hydraulic pressure from the backup pump to the NO. 1 and NO. 2 transfer modules, the second stage of the tail rotor servo, the rescue hoist, and the APU accumulator. A pressure switch on the module senses the backup pump operation and turns on the BACK-UP PUMP ON advisory light on the caution/advisory panel. If the flow rate through the module to the APU accumulator exceeds 1.5 gpm, a velocity fuse shuts off flow.

2.8.3 Primary Servo Shutoff

The purpose of the primary servo shutoff system is to allow the shutoff of an improperly operating servo system during flight and to test system operation before flight. The system is operated through a servo shutoff switch on each collective grip.

The system requires DC power for operation. The system is divided into a first-stage servo shutoff system and second-stage servo shutoff system. The first-stage system has a pressure switch on each first stage of the three primary servos. The same is true for the second-stage system. Each double-pole pressure switch has one switch for a caution light and another to route power to a servo system shutoff valve, when OFF is selected. The SERVO OFF switches on the pilot and ATO collectives do not control the tail rotor servos. The second stage tail rotor servo is controlled by a switch on the MISC SW panel marked TAIL SERVO, NORM, and BKUP, which should be used in the event of a mechanical failure of the first stage servo valve or linkage. With both servo systems pressurized, the pilot or ATO can shut down either servo stage. Due to an electrical interlock, both servo stages cannot be secured at the same time, nor can one be secured unless normal system conditions exist in the other. If the pilot selects the first-stage off, electrical power to the first-stage shutoff valve must pass through all three second-stage servo pressure switches to ensure the second stage is pressurized before shutdown can occur. As the first-stage pressure drops during shutoff, the first-stage pressure switches react to the decreasing pressure, causing the first-stage servo pressure caution light to illuminate and the control path for the second-stage shutoff to be interrupted. This prevents shutoff of the second stage during the time that the first stage is off. A selection of the second-stage off, in this example, would provide similar results. To close a servo shutoff valve, electrical power must be applied to the valve. When power is removed, the shutoff valve will open.

2.8.4 Hydraulic Leak-Detection/Isolation (LDI) System

The system protects the flight control hydraulic system by preventing loss of hydraulic fluid. The LDI system uses pressure switches and fluid level sensors for monitoring pump hydraulic fluid level, and pump pressure for primary, tail rotor, and pilot-assist servos. When a pump module reservoir fluid level switch detects a fluid loss, the logic module automatically operates the required shutoff valve(s) to isolate the leak and (with the exception of a leak at the pilot-assist servos) turns on the backup pump. In the cockpit, the RSVR LOW caution light for that system lights. Backup pump shutoff valve(s) operation is automatic through the logic module. If the leak continues, the pilot must place the SERVO switch to the appropriate OFF position. The heart of the LDI is composed of two logic modules.

2.8.4.1 Logic Modules

Two logic modules are used to control the operation of the hydraulic systems. The logic modules continually monitor the operation of the hydraulic systems by inputs received from pressure switches, fluid level switches, and control switch inputs. The outputs of the logic modules will turn on lights on the caution/advisory panel, notifying the pilot of a failure and/or turn off one or more valves due to a system malfunction to isolate a leak and maintain pressurization of the flight control system (Figures 2-27 and 2-28). All switching functions of the hydraulic logic modules are automatic, except as shown by Pilot Action on the logic diagrams.

2.8.4.2 Hydraulic Leak Test

The hydraulic leak test checks the complete hydraulic circuit with the exception of the transfer/shuttle valves. The HYD LEAK TEST switch is a three-position toggle switch located on the overhead console (Figure 1-6). The hydraulic leak test is a ground test and the following criteria must be achieved prior to initiating the test:

1. AC power.
2. Backup pump in AUTO position.
3. All hydraulic reservoirs full.
4. WOW.
5. Rotors turning.

The switch positions are TEST, RESET, and NORM. In the TEST position, if the above criteria are met, the test function activates an electrical signal illuminating the three RSVR low lights. The hydraulic logic modules then take their normal action for this condition. The following caution/advisory lights will illuminate, indicating a satisfactory test:

1. #1 TAIL RTR SERVO.
2. BOOST SERVO OFF.
3. SAS.
4. AFCS DEGRADED.
5. #1 RSVR LOW.
6. #2 RSVR LOW.
7. BACK-UP RSVR LOW.
8. BACK-UP PUMP ON.
9. #2 TAIL RTR SERVO ON.
10. MASTER CAUTION.

After a leak test has been made, the HYD LEAK TEST switch must be moved to RESET momentarily to turn off caution and advisory lights that were on during the test. Except for the HYD LEAK TEST switch, the hydraulic leak system consists of components of the NO. 1, NO. 2, and backup hydraulic systems. A WOW switch contact prevents hydraulic leak tests from being made in flight. Power to operate the hydraulic leak test system is from the NO. 2 DC primary bus through a circuit breaker located on the ATO circuit breaker panel, marked NO. 2 SERVO CONTR, and from the DC essential bus through a circuit breaker located on the overhead circuit breaker panel, marked BACKUP HYD CONTR.

2.8.4.3 Reservoir Fill System

A hand pump and manual selector valve are on the right side, upper deck of the helicopter for system servicing. The three main hydraulic system reservoir levels can be seen from the fill-pump location. The hand pump reservoir contains a sight gauge above the hand pump crank. A low-level mark indicates a requirement for refill. The hand pump reservoir serves as the rotor brake reservoir.

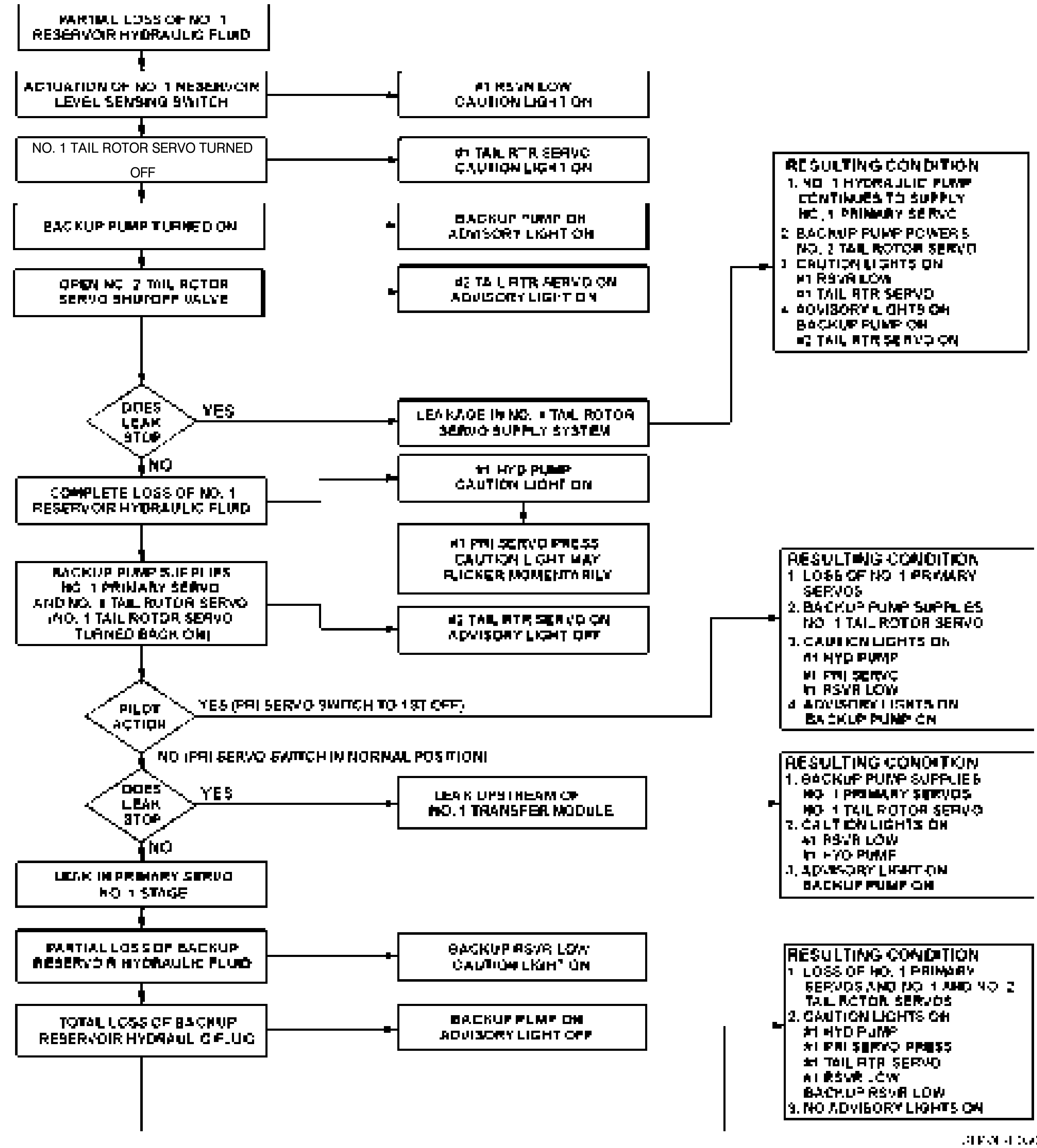


Figure 2-27. Hydraulic Logic Module Functions for a Leak in #1 Hydraulic System

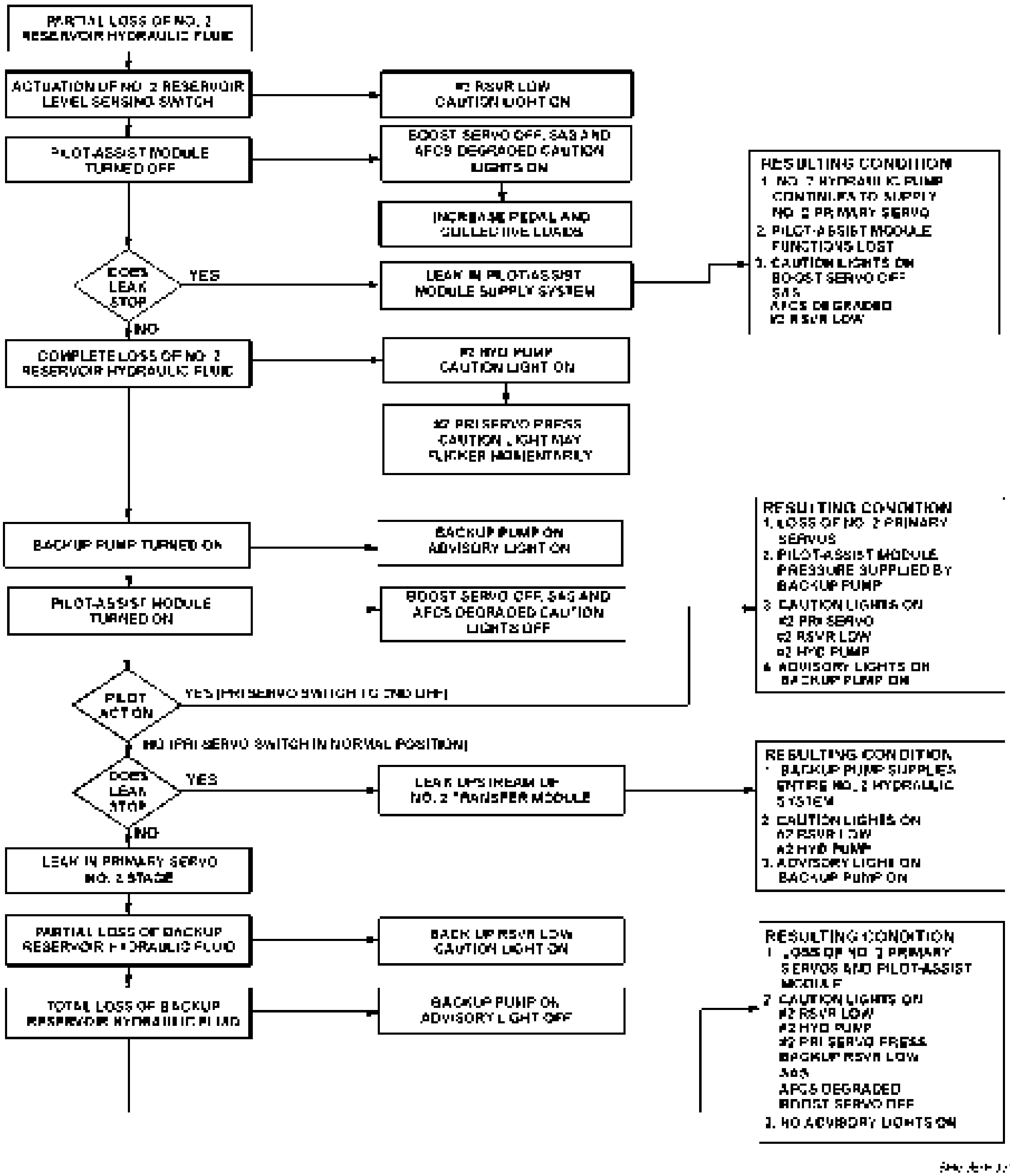


Figure 2-28. Hydraulic Logic Module Functions for a Leak in #2 Hydraulic System

2.9 FLIGHT CONTROL SYSTEM

The helicopter is controlled by varying and intermixing the control outputs from the cyclic, collective, and tail rotor pedals. Control movement is accomplished manually by either pilot or automatically by the AFCS. The flight control system can be divided into three sections:

1. Mechanical control system.
2. Flight control servo system.
3. Automatic flight control system

2.9.1 Mechanical Control System

The physical layout of the mechanical flight controls is presented in (Figure 2-29), and a simplified block diagram of the flight control system, including the AFCS components, is presented in (Figure 2-30). The cyclic, collective, and tail rotor pedal flight controls are routed aft and outboard of each pilot seat, vertically up each side of the aircraft, and are combined for each axis at the overhead torque shafts inside the hydraulics bay. The overhead torque shafts transfer inputs from the trim servos and flight controls through the pilot assist servos, and the mixing unit. From the mixing unit, fore, aft, and lateral inputs are transferred to the swashplate assembly via the primary servos and the bridge assembly. The yaw inputs to the tail rotor servo are transferred from the mixing unit aft to the tail rotor quadrant through the tail rotor cables.

Both pilot and copilot flight controls, control grips, and control grip function switches are identical and are presented in Figure 2-29. The copilot collective telescopes by twisting the grip and pushing the collective aft to improve access to the seat.

2.9.1.1 Tail Rotor Control System

The tail rotor control system provides directional control by varying the pitch of the tail rotor blades. The tail rotor servo is mechanically actuated, but requires hydraulic pressure to operate the pitch change shaft which moves the tail rotor pitch change beam, changing blade pitch angle through the pitch–change links.

The tail rotor servo is powered by either the NO. 1 hydraulic system or the backup hydraulic system. The tail rotor quadrant transmits tail rotor cable movement to the tail rotor servo. Two spring cylinders connected to the quadrant, allows cable tension to be maintained if either tail rotor cable becomes severed. Microswitches activate the TAIL ROTOR QUADRANT caution when either cable is broken. Directional control of the tail rotor is maintained by the remaining spring. If both cables are severed, two separate centering springs will counter the tail rotor servo pilot valve positioning the tail rotor to a neutral setting to provide a fly–home capability.

At the tail pylon fold area, the cables run though a series of pulleys that allow the tail pylon to be folded without disconnecting the control cables.

2.9.2 Flight Control Servo Systems

The flight control servo system (Figure 2-30) consists of the primary servos, tail rotor servos, and pilot–assist servos.

2.9.2.1 Primary Servos

There are three primary servos located in the hydraulics bay. Each primary servo has two stages that are independent and redundant with only the input linkage in common. Should one primary servo stage become inoperative due to pressure loss or a jammed input pilot valve, a bypass valve within the affected stage will automatically open, and the #1/#2 PRI SERVO PRESS caution will illuminate.

2.9.2.2 Tail Rotor Servo

The tail rotor servo has two independent stages. The first stage is powered by the NO. 1 hydraulic system with the TAIL SERVO switch in the NORM position. The second stage is powered by the backup hydraulic system with the TAIL SERVO switch in the BACKUP position. Should the first stage of the TAIL SERVO lose hydraulic pressure, the backup pump will automatically power the second stage of the tail rotor servo illuminating the #1 TAIL RTR SERVO caution, the #2 TAIL RTR SERVO ON advisory, and the BACKUP PUMP ON advisory.

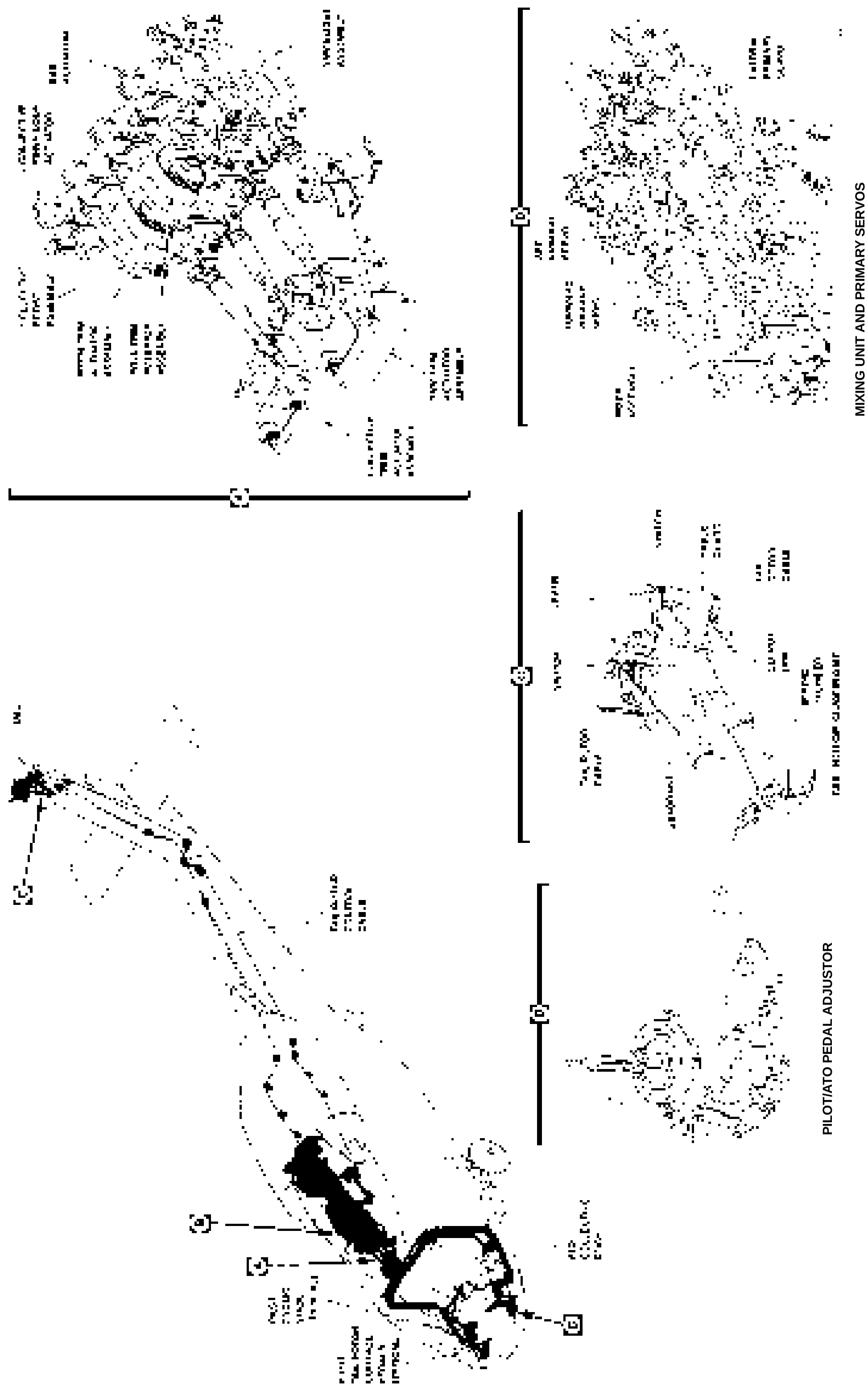


Figure 2-29. Mechanical Flight Controls

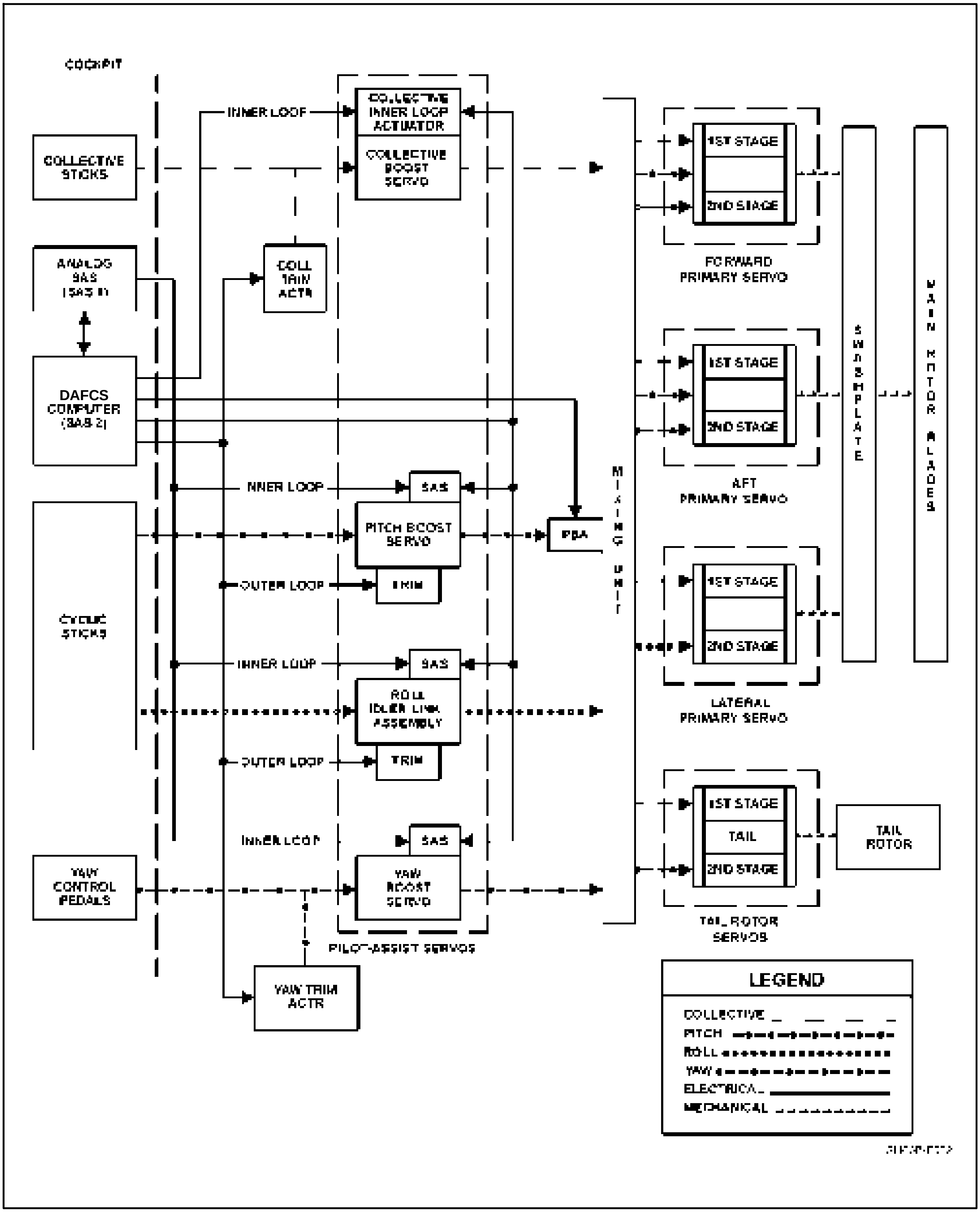


Figure 2-30. Flight Control System

2.9.2.3 Pilot Assist Servos

The pilot–assist servo assembly contains the boost servos, SAS actuators, and hydraulic (pitch and roll) trim actuators. Flight controls are operable without hydraulic pressure to the pilot–assist servos, but collective and yaw inputs will require considerable pilot effort. Hydraulic power is still required to move the primary servos.

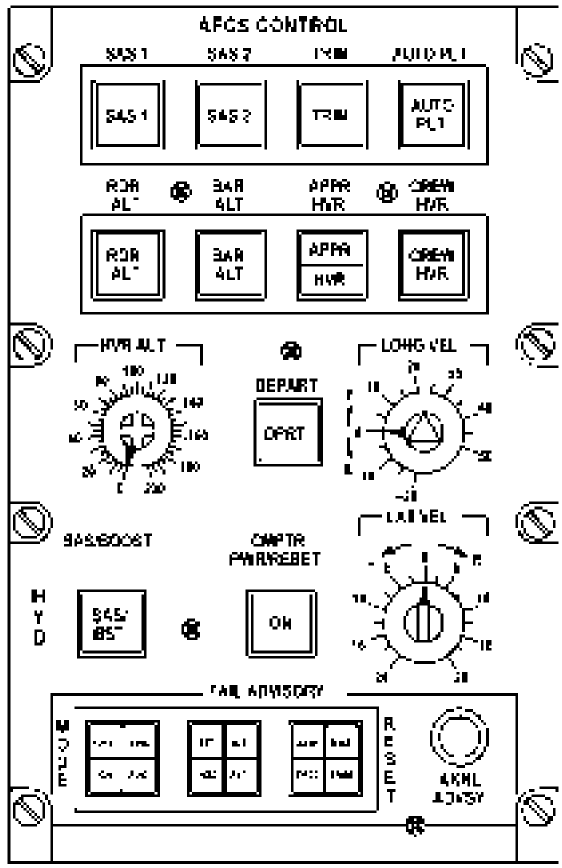
The pilot–assist servos, except pitch and roll trim, are turned on and off by pressing the SAS/BOOST pushbutton on the AFCS CONTROL panel (Figure 2-31). Hydraulic pressure to the pitch and roll trim actuators is turned on and off by the TRIM pushbutton on the AFCS CONTROL panel.

2.9.2.3.1 Boost Servos

There are three boost servos (collective, yaw, and pitch) located between the cockpit controls and the mixing unit, which reduce cockpit control forces and SAS system feedback. All boost servos and the roll channel incorporate a SAS actuator, which provides rate damping.

2.9.2.4 Control Mixing

The cyclic, collective, and pedal controls are mechanically combined in the mixing unit (Figure 2-32) to produce uncoupled airframe response characteristics. In addition to mechanical mixing, electrical (collective/airspeed–to–yaw) mixing is also present when trim is engaged.



CONTROL	FUNCTIONAL DESCRIPTION
SAS 1	Pushbutton that applies electrical power to the SAS 1 system. SAS 1 illuminates when system is on.
SAS 2	Pushbutton that applies electrical power to the SAS 2 system. SAS 2 illuminates when system is on.
TRIM	Pushbutton that applies electrical and/or hydraulic power to all control trim systems (pitch, roll, yaw, and collective). TRIM illuminates when trim systems are on.
AUTO PLT	Pushbutton that applies electrical power to the automatic pilot (SAS 1 or SAS 2 and TRIM must also be on). AUTO PLT illuminates when autopilot is on.
APPR/HVR	Pushbutton that engages either automatic approach or coupled hover mode, depending on flight regime.
CREW HVR	Pushbutton that activates crew hover mode while coupled hover mode is engaged.
DEPART	Light that illuminates when DEPART mode is selected via DEPART HOVER button on the cyclic grip.
LONG VEL	Rotary knob that permits the pilot to set a reference longitudinal velocity in coupled hover mode.
LAT VEL	Rotary knob that permits the pilot to set a reference lateral velocity during coupled hover mode.
CMPTR PWR/ RESET	Guarded pushbutton that applies electrical power to the AFCC. Also provides power on/reset capability if the computer automatically shuts down due to a self-diagnosed malfunction.
SAS/BOOST	Guarded pushbutton that applies hydraulic power to the SAS actuators and boost servos.
HVR ALT	Rotary knob that permits the pilot to set a reference hover altitude during automatic approach and coupled hover mode.
RDR ALT	Pushbutton that engages and disengages radar altitude hold system. It is illuminated when engaged and flashes when the collective TRIM RLSE switch is depressed. Reference altitude is obtained from HVR ALT rotary knob only when coupled hover mode is on. When coupled hover mode is off, the reference altitude is established when the pushbutton is engaged and reset when the collective trim switch is depressed and released.
BAR ALT	Pushbutton that engages and disengages barometric altitude hold. It is illuminated when engaged and flashes when the collective TRIM RLSE switch is depressed. Altitude is referenced automatically when the pushbutton is engaged and reset when collective TRIM RLSE switch is depressed and released.

Figure 2-31. Automatic Flight Control System (AFCS) Control Panel

NAME	COMPENSATION	CAUSE	COMPENSATION REQUIREMENT
MECHANICAL COMPENSATION			
Collective to yaw	Tail rotor thrust is increased	Main rotor torque	Nose yaws right when collective is increased
Collective to lateral	Rotor disc is tilted left	Lateral lead (tail rotor propeller effect)	Helicopter drifts right when collective is increased
Collective to longitudinal	Rotor disc is tilted forward	Rotor downwash on stabilator	Nose pitches up and helicopter drifts aft when collective is increased
Yaw to longitudinal	Rotor disc is tilted aft	Tail rotor lift vector	Nose pitches down and helicopter drifts forward when left pedal is applied
ELECTRONIC COMPENSATION			
Collective/airspeed to yaw	A portion of the main rotor torque compensation is provided by a trim input that is proportional to collective position and airspeed. The trim input is then progressively washed out as pylon side loads increase with airspeed.	Camber of tail rotor pylon varies side load with airspeed	Nose yaws left as airspeed increases

Figure 2-32. Control Mixing

2.9.3 Automatic Flight Control System (AFCS)

The AFCS is an electrohydromechanical system which provides inputs to the flight control system to assist the pilot in maneuvering and handling the helicopter. The AFCS is composed of three major subsystems: the SAS, the stabilator system, and the DAFCS. A schematic of the AFCS is presented in Figure 2-33. All engagement controls for the three subsystems are contained on the AFCS and stabilator control panels. Each subsystem operates independently of the other two subsystems and they all complement one another. Autopilot functions are engaged by the AUTO PLT pushbutton on the AFCS CONTROL panel. The AFCS RELEASE pushbutton on the cyclic will disengage SAS 1, SAS 2, and autopilot. The AFCS provides the following features:

1. Pitch, roll, and yaw stability augmentation.
2. Stabilator control.
3. Cyclic, collective, and pedal trim.
4. Pitch and roll attitude hold.
5. Airspeed hold.
6. Heading hold.
7. Barometric altitude hold.
8. Radar altitude hold.
9. Pitch and roll hover augmentation/gust alleviation.
10. Turn coordination.
11. Maneuvering stability.
12. Automatic approach to hover.
13. Hover coupler.
14. Automatic depart.

15. Crew hover.
16. Longitudinal stick gradient augmentation (pitch bias actuator).
17. Blade fold assist.
18. Automatic preflight check.
19. Diagnostics (mode failure display).

2.9.3.1 AFCS Control Panels

The AFCS is controlled from two panels: the AFCS CONTROL panel and the STABILATOR control panel. Both control panels are located on the lower console. Controls and functions of the AFCS CONTROL panel are summarized in Figure 2-31. Functional descriptions of the fail advisory lights displayed on the AFCS CONTROL panel are presented in Figure 2-34. The STABILATOR control panel contains all the operating controls for the stabilator and is shown in Figure 2-36. All the other AFCS controls are on the AFCS CONTROL panel. All detectable AFCS mode failures except for the stabilator illuminate the AFCS DEGRADED light on the caution/advisory panel and the appropriate mode failure capsule on the FAILURE ADVISORY section of the AFCS CONTROL panel. Stabilator failures illuminate the STABILATOR caution light on the caution/advisory panel and generate an aural warning tone in the pilot and ATO headsets.

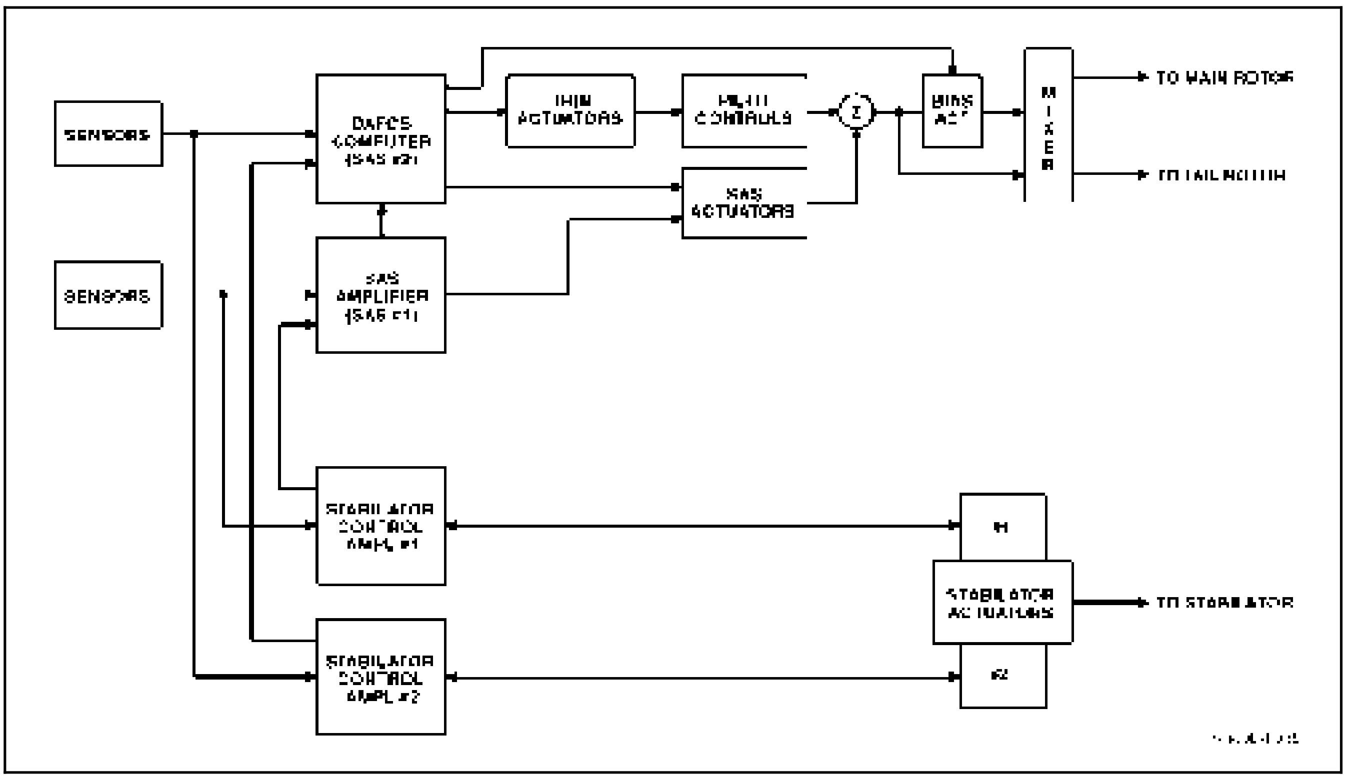


Figure 2-33. AFCS Input, Simplified Block Diagram

FAIL ADVISORY LIGHT ON	SYSTEM	FUNCTIONAL DESCRIPTION
CPLR	Approach/Hover Coupler	Approach/hover coupler capability loss. ATT, or A/S, or ALT light may also be on.
CORD	Turn Coordination	Turn coordination is lost if AUG or SAS 1 and SAS 2 or A/S light is on.
CH	Crew Hover	Loss of crew hover mode.
AUG	Longitudinal and Lateral Hover Augmentation	Loss of hover augmentation. A/S, ATT, or SAS 1, and SAS 2 light may also be on.
ATT	Pitch and Roll Attitude Hold	Pitch or roll autopilot attitude hold has been lost. If TRIM is also on, both pitch and roll attitude hold have been lost or malfunctioning. If pitch attitude failure occurs, the A/S light also goes on, indicating airspeed hold is also not available.
A/S	Airspeed Hold	Airspeed hold is lost. The autopilot may continue to hold pitch attitude unless a failure of pitch attitude hold occurs. In this case, ATT light would go on.
HDG	Heading Hold	Heading hold has been lost or the ability to synchronize heading hold has been lost, or collective to yaw electrical coupling has been lost.
ALT	Barometric or Radar Altitude Hold	If barometric altitude hold was engaged, altitude hold is lost. If radar altitude hold is desired, it must be selected. If radar altitude hold was engaged, barometric altitude hold will automatically engage, unless the type of failure prevents any altitude hold operation.
SAS 1	SAS 1 Pitch, SAS 1 Roll, SAS 1 Yaw	A malfunction has occurred, causing improper pitch, roll, or yaw SAS 1 operation. SAS 1 should be disengaged to ensure proper SAS 2 operation.
BIAS	Pitch Bias	Loss or modified operation of the pitch bias actuator occurs, if the BIAS light goes on. An airspeed transducer failure causes the pitch bias actuator to move to a position corresponding to 120 KIAS. The bias actuator continues to function about this 120 knot airspeed reference. The A/S light should also be on if airspeed is the cause of the BIAS light. A vertical gyro (pitch) failure causes the bias actuator to move to a center position. The bias actuator now stays at this center position. The ATT light should also be on, if attitude is cause of bias light. A dual pitch rate gyro failure will cause the bias actuator to lose its ability. However, the bias actuator will continue to function for attitude and airspeed inputs. The SAS 1 and SAS 2 lights should also be on if the pitch rate gyros are the cause of failure. The pitch bias actuator itself is not responding to computer commands. The bias actuator stops wherever it is at that time. No other fail advisory lights are coupled with the BIAS light for this problem.

Figure 2-34. Fail Advisory Light (Sheet 1 of 2)

FAIL ADVISORY LIGHT ON	SYSTEM	FUNCTIONAL DESCRIPTION
SAS 2	SAS 2 Pitch, SAS 2 Roll, SAS 2 Yaw	Any combination of SAS 2 axis failure causes the affected axis to be automatically disabled. If SAS 2 and SAS 1 fail advisory lights go on, turn off SAS 1.
TRIM	Pitch Trim Actuator, Roll Trim Actuator, Yaw Trim Actuator, Collective Trim Actuator	Cyclic pitch or roll trim, yaw pedal trim, or collective trim is lost or malfunctioning. Any combination of the above will cause the TRIM light to go on.
No fail advisory light on, AFCS DEGRADED caution light flashing	Unknown	A degradation has occurred that is not associated with a fail advisory light.
No fail advisory light on, AFCS DEGRADED caution light on steady	Primary power to the computer	A computer power sever has occurred. All DAFCS computer output signals are discontinued. To attempt to regain computer operation, press CMPTR PWR/RESET switch OFF then ON. Loss of all AFCS functions except SAS 1, boost, and Stabilator.

Note

- Use the AKNL ADVSY switch to acknowledge the failure of an AFCS system, so that MASTER CAUTION light will be reset for a subsequent failure.
- To reset the computer for any mode failure (except a power sever), press any one of the three FAIL ADVISORY MODE RESET switches.
- If the CMPTR PWR/RESET switch is on, and SAS 2 is off, pressing any of the three FAIL ADVISORY MODE RESET switches will cause SAS 2 to automatically go on. This is a backup means of engaging SAS.

Figure 2-34. Fail Advisory Light (Sheet 2)

2.9.3.2 DAFCS

The central component of the DAFCS is the digital computer. The computer commands the pitch bias actuator (PBA), the inner-loop SAS actuators, and the outer-loop trim actuators in all four control channels. A system block diagram of the DAFCS is presented in Figure 2-35. The computer also provides self-monitoring, fault isolation, and failure advisory.

The DAFCS employs two types of control, identified as inner loop and outer loop. The inner loop (SAS) employs rate damping to improve helicopter stability. This system is fast in response, limited in authority, and operates without causing movement of the flight controls.

The outer loop (AUTOPILOT) provides long-term inputs by trimming the flight controls to the position required to maintain the selected flight regime. It is capable of driving the flight controls throughout their full range of travel (100 percent authority) at a limited rate of 10 percent per second. Both inner and outer loops allow for complete pilot override through the normal use of the flight controls.

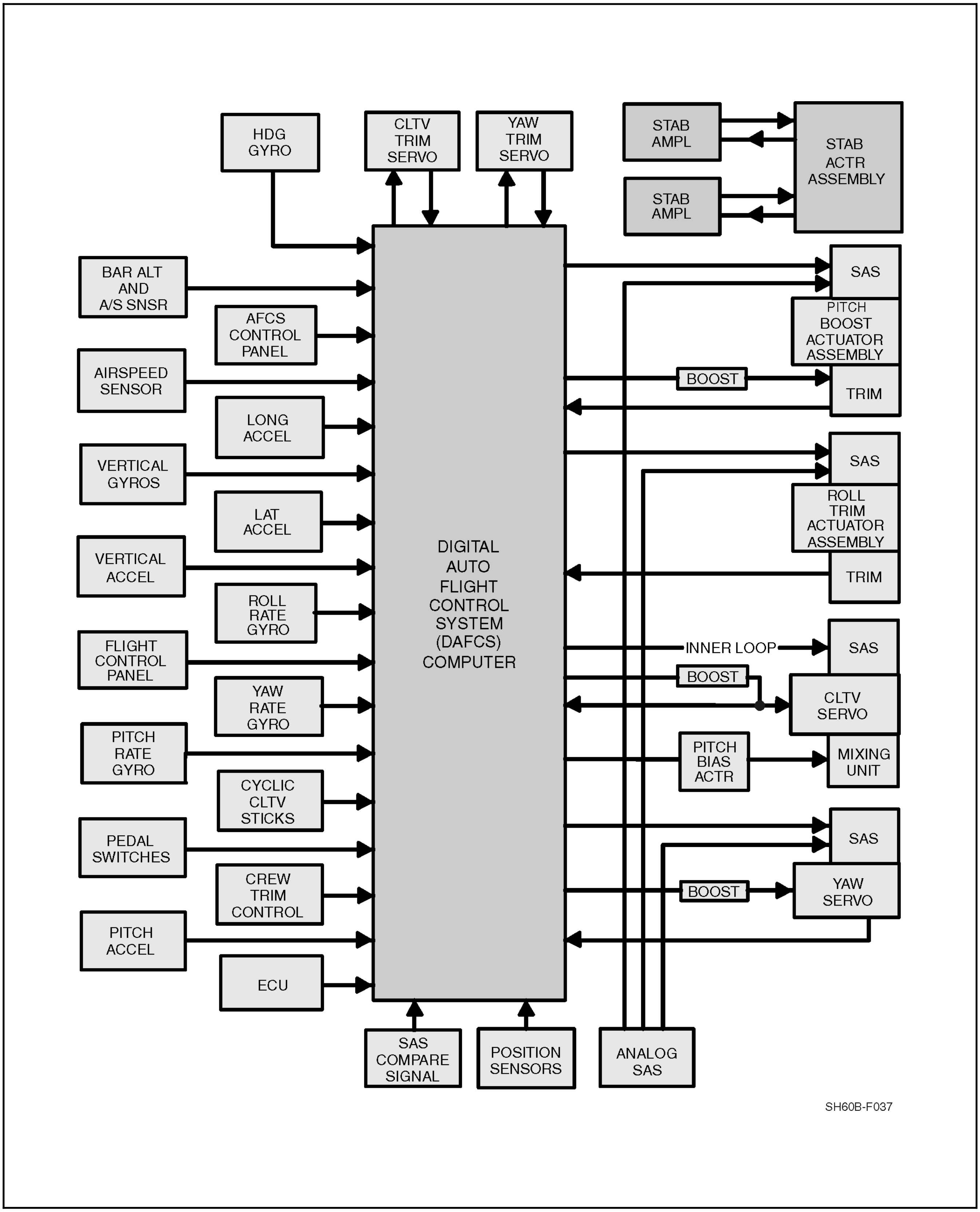


Figure 2-35. DAFCS Input/Output Block Diagram

WARNING

- When AFCS computer power is cycled, trim is disengaged and an unguarded cyclic will allow the rotor arc to dip as low as four feet above the deck, prior to full control deflection, without pounding the droop stops.
- AFCS computer power is interrupted with a total AC power failure. After AC power is restored, AFCS computer power must be cycled before resetting AFCS functions.

The DAFCS computer processes incoming information from various sensors aboard the aircraft and stores this information in its memory. The sensor information is used by the computer central processing unit (CPU) to compute required correction signals. Inner-loop correction signals are routed to the SAS actuators and outer-loop signals are routed to trim servos and actuators.

2.9.3.2.1 AFCS Voltage Sensor Relay

The function of the voltage sensor relay (VSR) is to select the source of three-phase, 115 Vac power to be applied to the DAFCS computer. The voltage sensor also supplies power to the Heading Attitude Reference System (HARS) via the ac essential bus relay. Power source selection is a function of the NO. 1 generator output integrity. When 3 phase power from the NO. 1 ac primary bus is sensed in the voltage sensor, the ac essential bus relay is energized and ac power is routed to the AFCS power switching assembly via the VSR.

When there is a phase disruption or fault occurrence on the NO. 1 ac primary bus supply, the VSR will automatically switch the power source of the AC Essential bus from the NO. 1 ac primary bus to the NO. 2 ac primary bus to retain basic HARS, SAS and TRIM functions. If the VSR switches power to the NO.2 AC primary bus, the DAFCS could fail and the AFCS, SAS and TRIM functions may be required to be reset. A malfunctioning VSR may cause a power failure of the DAFCS computer, multiple fail advisory lights, tumbling or precessing of either or both pilot or ATO AI or BDHI, or steady or intermittent OFF flags presented in the pilot or ATO AI or BDHI.

The AFCS voltage sensor is mounted overhead in the forward cabin area. Power to the VSR is supplied from both the NO. 1 ac primary bus through the AFCS CMPTR CB on the center circuit breaker panel and from the NO. 2 Primary bus through the AFCS CMPTR CB on the sensor operator circuit breaker panel. The voltage sensor for the VSR is the circuit breaker labeled AC ESNTL BUS SUPPLY (7.5 amp) on the NO. 1 AC Primary bus located on the center circuit breaker panel. Pulling the VSR CB (i.e. AC ESNTL BUS SUPPLY CB) will cause power to both the DAFCS and the AC Essential bus to be supplied from the NO. 2 AC Primary side. Caution should be taken to ensure the AC ESNTL BUS SUPPLY CB on the NO. 2 Primary Side located on the Corner Circuit Breaker Panel is pressed in before pulling the VSR CB.

2.9.3.3 Stabilator System

The stabilator system optimizes trim attitudes for cruise, climb, and autorotation and provides pitch stability augmentation to complement the SAS for additional redundancy. The stabilator system is completely independent of the other two AFCS subsystems except for common airspeed sensors, lateral accelerometers, and pitch rate gyros. The stabilator control system is a completely automatic fly-by-wire control system with a manual backup slew control. The primary purpose of the stabilator control system is to eliminate undesirable noseup attitudes caused by rotor downwash impinging on the horizontal stabilator during transition to a hover and during low-speed flight. The stabilator panel (Figure 2-36) contains an automatic control (AUTO CONTROL) switch, a TEST pushbutton, and a manual slew (MAN SLEW) switch. The AUTO CONTROL switch can only be used to engage the automatic mode or to reset the stabilator in the event of a stabilator failure. The switch is not an alternate action pushbutton. The pilot can manually position the stabilator to any position within the stabilator limits by moving the MAN SLEW switch. The TEST pushbutton, which is operational below 50 knots, is used to check the automatic mode fault detector. The stabilator is positioned by two electric jackscrew actuators acting in series. Each actuator provides one-half the input to position the stabilator and is controlled by a separate and redundant stabilator amplifier. The stabilator travels from 42° trailing edge down for hover and low–speed flight below 30 knots to 10° trailing edge up for cruise and maneuvering flight. The stabilator electrical screw actuators receive power from the DC essential bus and NO. 1 DC primary bus through two circuit breakers marked STAB SYS PWR and STAB PWR, respectively. The circuit breakers are located on the overhead console and ATO circuit breaker panels, respectively. Four inputs are required to position the stabilator (Figure 2-37):

1. Airspeed.
2. Collective position.
3. Lateral acceleration.
4. Pitch rate.

The airspeed input aligns the stabilator with the main rotor downwash during slow–speed flight. The collective position input decouples aircraft pitch attitude with collective position. Pitch rate and lateral acceleration inputs improve the dynamic response of the aircraft, especially in gusty air conditions. The pitch rate input supplements the dynamic stability provided by the SAS and DAFCS, and the lateral accelerometer input decouples the aircraft pitch response with changes in the rotor downwash on the stabilator during out of balanced flight.

Each stabilator amplifier receives these four inputs but receives the inputs from independent sensors. The DAFCS computer monitors each of these sensors for malfunctions, and the stabilator control system monitors and compares the position of the two actuators. Any system malfunction caused by a difference between the two stabilator actuator positions results in stabilator remaining in the last position, an automatic power shutdown to both actuators, an aural tone to the pilots, and a STABILATOR caution light on the caution/advisory panel.

WARNING

It is possible for the stabilator to fail without illumination of the STABILATOR caution light and associated aural warning tone. In this case, the first indication of failure will be an uncommanded pitch change.

If a malfunction of the stabilator system occurs, the pilot has the ability to manually position the stabilator with the manual slew switch on the stabilator control panel. The manual slew switch bypasses the stabilator amplifier automatic mode and applies power directly to the actuators through relays in the amplifiers. A stabilator position indicator aids the pilot in positioning the stabilator to any position between the stabilator travel limits. However, the total travel is restricted if the malfunction is caused by an actuator failure. The stabilator travel is restricted to 35° if an actuator fails in the full–down position or 30° if an actuator fails in the full–up position. The stabilator control rate is limited to ±6° per second.

WARNING

Reengagement of the automatic mode after a shutdown occurs results in the automatic mode operating for one second. If a hardover signal to one actuator was the cause of the initial shutdown, and reengagement is attempted, that actuator will again cause the stabilator to move before another disengagement is commanded. In this case subsequent reengagement shall not be attempted since it may result in additional stabilator movement.

Note

Due to the loss of automatic fail safe features with the stabilator in the manual mode, intentional flight in the manual mode is not recommended.

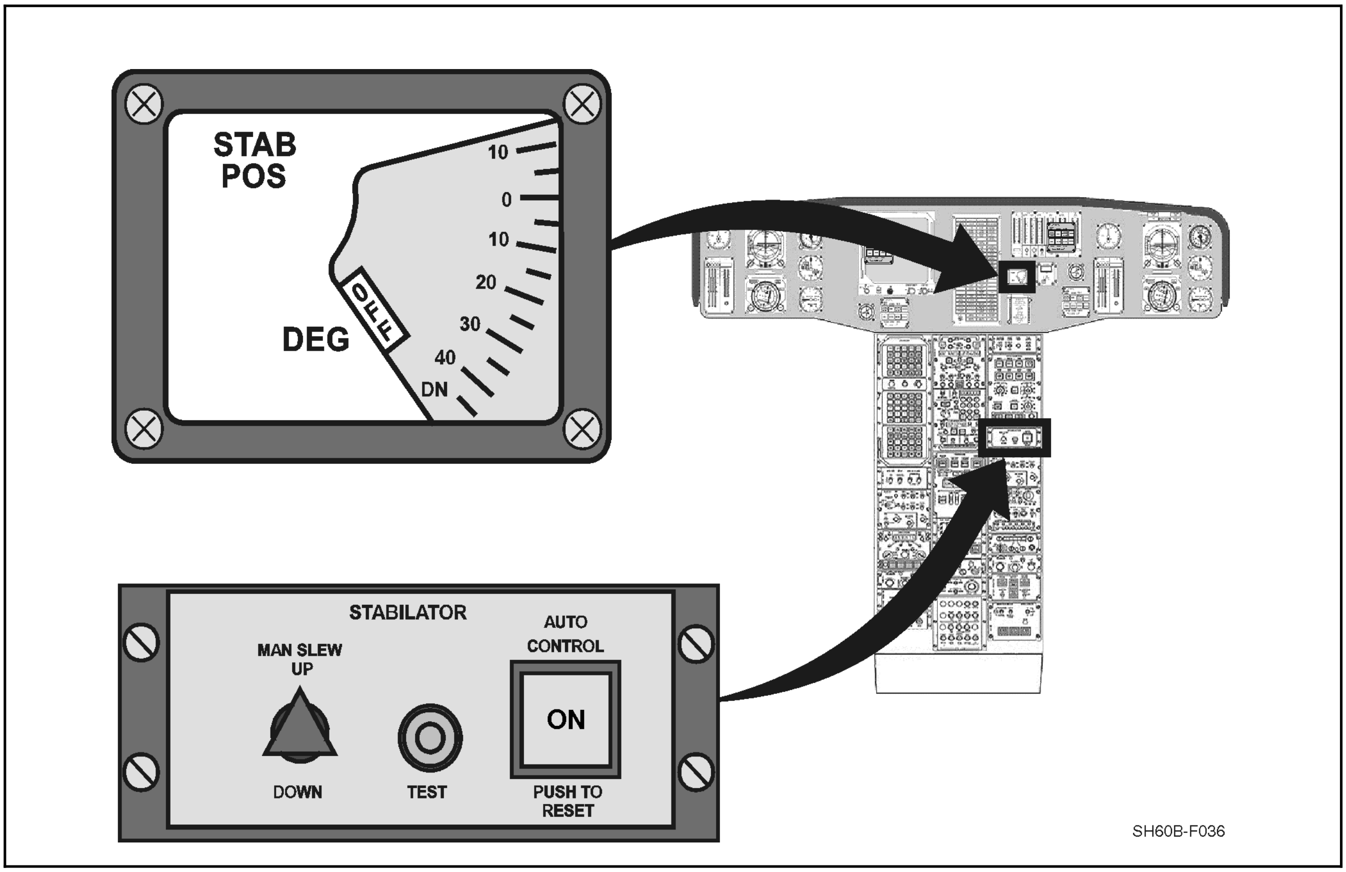


Figure 2-36. Stabilator Control Panel

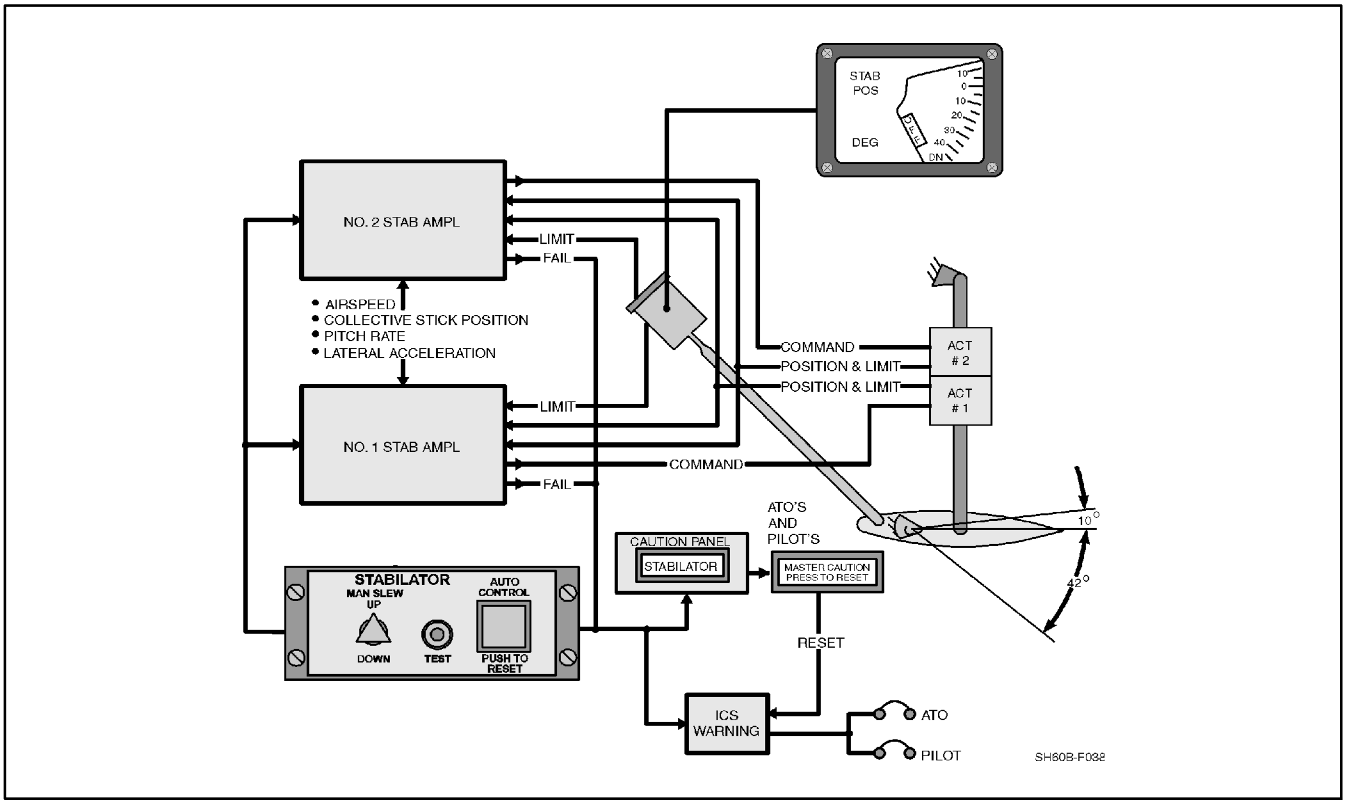


Figure 2-37. Stabilator Block Diagram

2.9.4 Stability Augmentation System

SAS provides improved stability by sensing acceleration in the pitch, roll, yaw and vertical axes, and by applying a control input to stop the acceleration and maintain a constant rate (Figure 2-38). The SAS is an inner-loop system with two separate and independent SAS channels. SAS channel NO. 1 is an analog system; SAS channel NO. 2 is a digital system and is part of the DAFCS computer. Both SAS channel functions are identical except for the hover augmentation/gust alleviation and hover coupler DAFCS features, which are incorporated only into SAS 2. SAS 2 also complements the DAFCS to provide turn coordination and roll attitude hold. With both channels engaged, the pitch, roll, and yaw SAS actuators have ±10 percent control authority with each channel providing ±5 percent.

Only SAS 2 commands the collective SAS actuator. The collective SAS only operates in RDR ALT, BAR ALT, APPR HVR, and DEPART modes and is limited to ±10 percent control authority. A system block diagram of the SAS is presented in (Figure 2-38). Hydraulic pressure to the SAS actuators is turned on by the SAS/BOOST switch on the AFCS CONTROL panel. This controls power to the SAS shutoff valve on the pilot assist module. A loss of SAS actuator pressure is monitored by the SAS pressure switch which lights the SAS caution light. Either SAS 1 or SAS 2 may be operated separately or simultaneously.

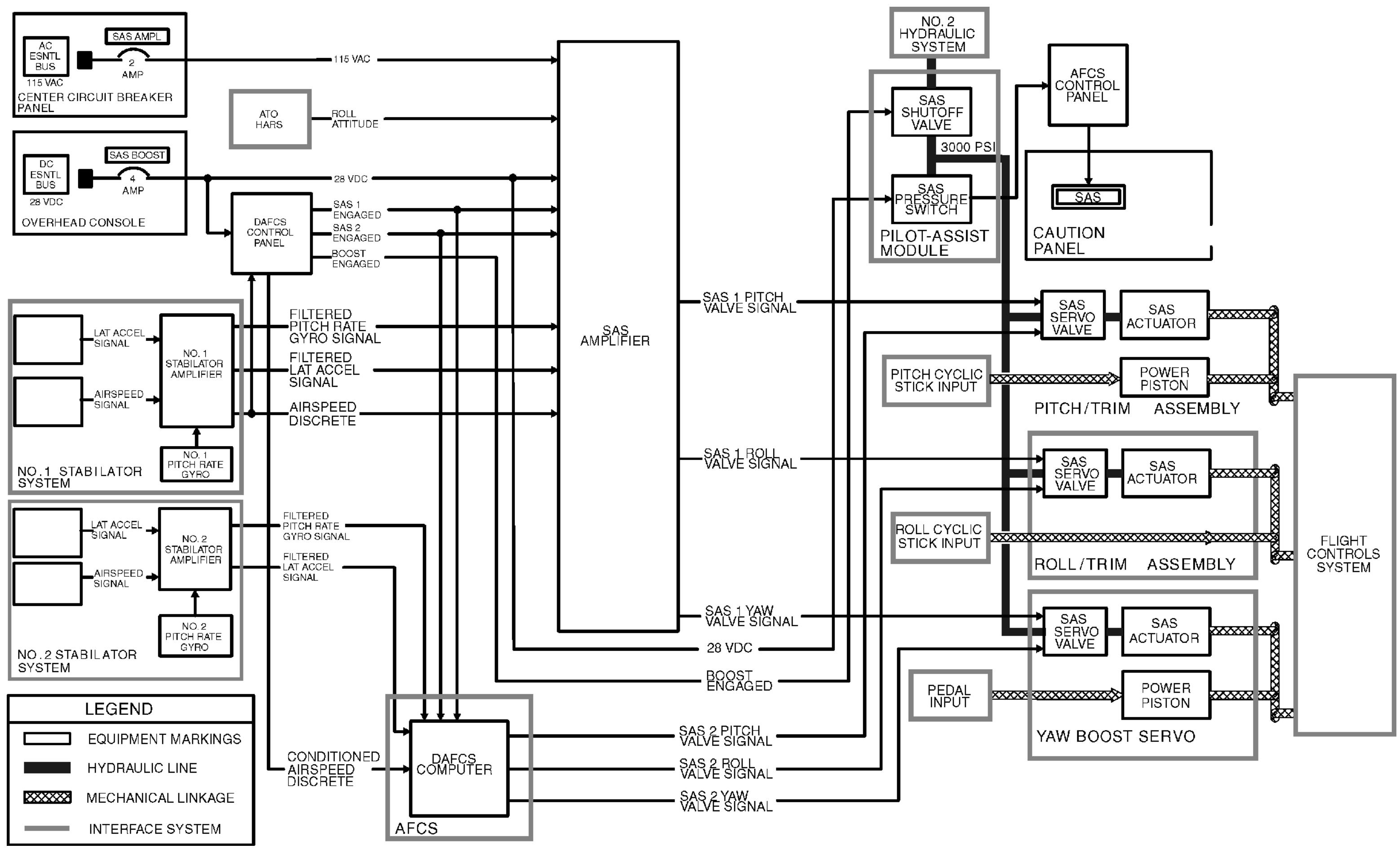


Figure 2-38. Flight Control Hydraulic System (Sheet 1 of 2)

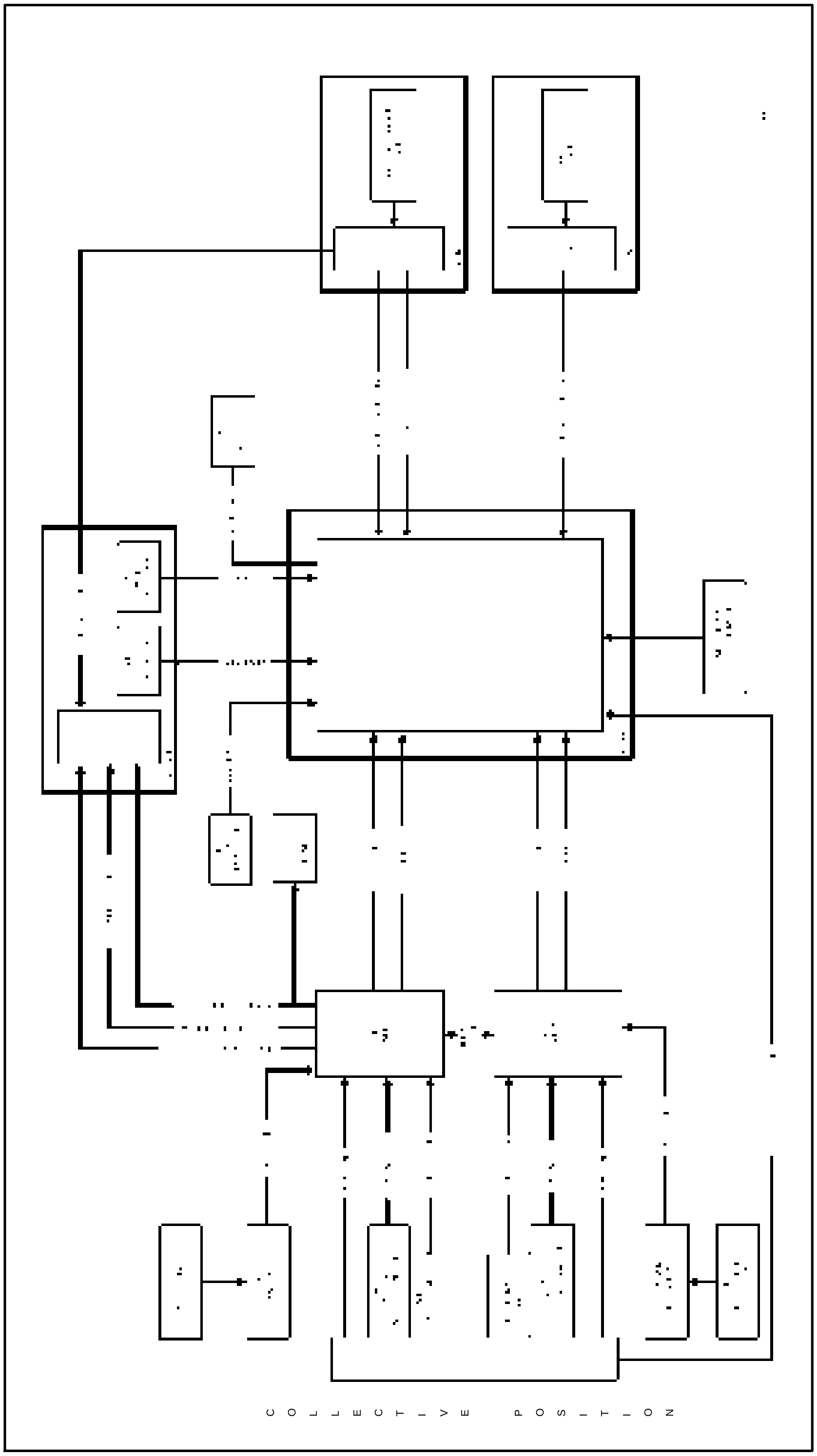


Figure 2-38 Flight Control Hydraulic System (Sheet 2)

For each control axis (except vertical), command signals from both SAS channels are applied simultaneously to separate coils of an electrohydraulic servo valve. The two signals are summed to provide a single input into the flight control system through a single series SAS actuator. The operation of the SAS channels is continuously monitored by the DAFCS. If either of the two SAS channels malfunctions, the AFCS DEGRADED caution light on the caution/advisory panel flashes and the appropriate SAS fail advisory light on the AFCS CONTROL panel is illuminated. If SAS 2 has failed, the computer will automatically disengage the affected axis. If SAS 1 has failed, the pilot must disengage the failed SAS 1 channel from the AFCS CONTROL panel. The remaining SAS channel is limited to ±5 percent authority but operates at twice its normal gain to partially compensate for the failed SAS channel. Either SAS channel can be disengaged by pressing the appropriate SAS button on the AFCS CONTROL panel. Both SAS channels are disengaged by pressing the AFCS REL button on the cyclic. If both SAS 1 and SAS 2 lose power, are manually shut down, or hydraulic pressure is lost to the actuators, the SAS caution light will illuminate.

2.9.4.1 Trim System

The parallel trim actuator assemblies provide the flight control force gradients and detent positions and the outer-loop autopilot control functions. The trim actuators command full control authority in all four control channels but are rate-limited to 10 percent per second. Pressing TRIM REL switch (cyclic trim release, collective trim release, pedal release) disengages the respective trim function and allows free control motion. Releasing trim release switch reengages trim if the cyclic stick is not moving. For yaw trim release above 50 knots, the pedal microswitches and the cyclic trim switch must be pressed. Below 50 knots, only the pedal microswitches have to be pressed. If the trim system fails, the TRIM fail advisory light on the AFCS CONTROL panel will illuminate and the pilot can compensate for the failure. The pilot is able to override the trim control forces in all channels.

2.9.4.2 Autopilot

The autopilot maintains helicopter pitch and roll attitude, airspeed, and heading during cruise flight, provides maneuvering stability, and a coordinated turn feature at airspeeds above 50 KIAS. The autopilot function is engaged by pressing the control panel SAS 1 or SAS 2 switches, TRIM switch, and then pressing the control panel AUTO PLT pushbutton. The autopilot may be disengaged by pressing the AUTO PLT pushbutton on the AFCS panel or pressing the AFCS REL button on the cyclic. When engaged, 28 Vdc is supplied from the control panel to the computer. The computer also provides command signals to the trim actuators to reposition the flight controls using the trim system. With SAS 2, TRIM, and autopilot on, all DAFCS functions are available. With SAS 1, TRIM and autopilot on, all functions remain available except Hover Augmentation/Gust Alleviation and “Collective SAS” functions (i.e., coupled approach, altitude hold).

2.9.4.3 Attitude and Airspeed Hold

Attitude and airspeed hold are engaged with AUTO PLT. In the pitch channel, at airspeeds less than 50 knots, attitude changes are commanded by changing the cyclic position with the TRIM REL switch or use of the four-direction (beeper) TRIM switch. This causes the cyclic to move and the helicopter attitude to change approximately 5° per second. When cyclic movement is stopped, the autopilot stabilizes the helicopter around the new cyclic position and attitude. Above 50 knots and bank angles less than 30°, the system becomes airspeed sensitive in pitch. Actuating the four-direction TRIM switch will cause the cyclic to move and the helicopter to change airspeed reference at the rate of 6 knots per second. Because of variations in pitot-static systems during gusty conditions, an integrated longitudinal acceleration input is used for short-term corrections. The airspeed sensor is used for long-term updates through a 3 second filter. The roll channel autopilot holds roll attitude of the helicopter. Attitude information is supplied to the computer from the pilot and ATO A/A24G vertical gyros. The command signal is applied to roll SAS 1 and SAS 2 and the roll trim system. When the pilot actuates the four-direction TRIM switch, the helicopter roll attitude will change at approximately 6° per second. In addition to the attitude hold feature, an automatic wing-leveling capability is also included. During transitions from hover to airspeeds above 50 knots, this feature automatically retrims the aircraft from a left roll attitude in a hover to a wings level attitude at 50 knots. Once a level attitude is established, the attitude hold feature maintains that attitude until a new roll attitude is commanded by the pilot.

WARNING

Uninterrupted use of the four–way cyclic TRIM switch to increase speed may cause the aircraft to enter a descent with altitude hold engaged without an associated caution light. Manual input is required to arrest the descent.

2.9.4.4 Heading Hold

The yaw channel of the autopilot provides the heading hold feature for hover and forward flight and is engaged whenever the AUTO PLT pushbutton switch is illuminated. Heading hold is an outer–loop function operating through the yaw trim actuator, and therefore will only be operational whenever the yaw trim is engaged. Releasing all pedal switches at a given heading synchronizes the trim system to the established heading. A potentiometer in the yaw trim actuator applies a trim position feedback signal to the computer and then cancels the drive signal at the desired position, stopping the motor. The yaw autopilot also uses a collective position sensor to hold reference heading for yaw excursions caused by main rotor torque changes. The collective position sensor is controlled by an airspeed signal which reduces its gain as airspeed increases. When heading hold is engaged, the HDG TRIM (slew) switch on the collective allows the pilot to make heading changes without retrimming. Below 50 KIAS, the aircraft is slewed at 3° per second. Above 50 KIAS, actuation of the switch.

for less than 1 second provides a 1° heading change and actuation for greater than 1 second provides a 1° per second coordinated turn. The heading hold is reengaged following a turn when the following conditions are maintained for 2 seconds:

1. Aircraft roll attitude is within 2° of wings level.
2. Yaw rate is less than 2° per second.

The heading hold is disengaged by the WOW switch when the aircraft is on the ground.

2.9.4.5 Altitude Hold

Either barometric or radar altitude hold is selectable from the AFCS CONTROL panel (but not simultaneously). When the altitude hold mode is selected, the DAFCS computer uses as a reference altitude the existing altitude from either the air data transducer if barometric altitude hold is selected or the radar altimeter if the radar altitude hold is selected. The computer commands both the collective SAS actuator and the collective trim actuator to maintain the reference altitude. The SAS actuator provides fast–response, limited–authority corrections, and the trim actuator provides limited–response (rate limited), full–authority corrections. The DAFCS computer uses altitude and rate from the barometric or radar altitude systems (depending on which hold mode is selected) and vertical acceleration to command the collective SAS and trim actuators. The computer also monitors engine torque to prevent dual–engine torque from exceeding 116 percent whenever the collective trim is positioning the collective. Barometric altitude hold is engaged at any altitude and airspeed by depressing the BAR ALT pushbutton switch with SAS 2 and autopilot engaged. Depressing the collective TRIM REL button temporarily disengages the mode. Upon release of the trim switch, barometer altitude hold automatically reengages and maintains the altitude at the time of reengagement. Radar altitude hold is engaged at any altitude from 0 to 5,000 feet AGL and at any airspeed by depressing the RDR ALT pushbutton with SAS 2 and autopilot engaged.

When in the hover coupler mode, altitude hold is referenced to the altitude selected on the AFCS CONTROL panel HVR ALT potentiometer. Depressing collective TRIM REL temporarily disengages the mode. Upon release of the trim switch, radar altitude hold automatically reengages to the altitude selected on the AFCS CONTROL panel HVR ALT potentiometer. When in the hover coupler mode, transition from one altitude to another is made with the HVR ALT knob on the AFCS CONTROL panel. Resulting climb/descent rates are limited to 1,000/500 feet per minute, respectively. If the radar altitude mode fails while engaged, barometric altitude hold is automatically engaged.

2.9.4.6 Hover Augmentation/Gust Alleviation

An additional feature of SAS, provided only through SAS 2, is hover augmentation/gust alleviation. It further improves aircraft stability at low airspeed using attitude retention and longitudinal and lateral acceleration to eliminate drift.

2.9.4.7 Turn Coordination

Automatic turn coordination is provided at airspeeds greater than 50 knots. Turn coordination allows the pilot to fly a coordinated turn with directional control provided by the AFCS. The AFCS uses lateral acceleration and roll rate to determine if the aircraft is out of balanced flight and provides the yaw SAS and yaw trim with the inputs necessary to maintain an automatic coordinated turn. Automatic turn coordination is engaged and heading hold disengaged when roll attitude is greater than 1° and any of the following conditions exists:

1. Lateral cyclic force greater than 3.0 percent of cyclic displacement.
2. Cyclic trim release is pressed.
3. Roll attitude is beeped beyond 2.5° bank angle.

2.9.4.8 Maneuvering Stability

Pitch control forces are increased to increase pilot effort required for a given pitch rate at bank angles greater than 30°. The higher pitch control forces help alert the pilot to G–loading during maneuvering flight and are provided through the longitudinal trim actuator. A linear longitudinal stick force gradient is provided by trimming 1 percent forward stick for each 1.5° angle of bank between 30° and 75°. At 75° angle of bank, the longitudinal stick force is equivalent to 30 percent of stick displacement. The maneuvering stability feature is engaged whenever the AUTO PLT pushbutton on the AFCS CONTROL panel is illuminated.

2.9.4.9 Auto Approach to a Hover

An automatic approach can be initiated from any airspeed and any altitude below 5,000 feet AGL. The approach should be initiated from level flight into the wind. If the approach is initiated in a banking turn, the AFCS will make a spiraling approach. Prior to the helicopter slowing to 60 KIAS, the pilot should level the wings and establish the desired heading.

WARNING

> Initiating an automatic approach while in a trimmed turn may result in a spiraling approach which will continue through the selected altitude. Immediate pilot action will be required to avoid water impact.

The DAFCS provides the capability to perform an automatic approach to a zero longitudinal and any lateral groundspeed selected on the LAT VEL control knob on the AFCS CONTROL panel and to any radar altitude selected on the HVR ALT control knob, between 40 feet and 200 feet. If the HVR ALT is set below 40 feet, the approach will be made to 40 feet and then continued to the HVR ALT setting when the mode is switched from APPR to HVR. The helicopter will be commanded to the LONG VEL setting of the control when the mode is switched from APPR to HVR. The automatic approach can be initiated with SAS 2, TRIM, and AUTO PLT engaged by activating the automatic approach pushbutton (APPR/HVR) on the AFCS CONTROL panel. The automatic approach is an outer loop only function and commands the aircraft to decelerate or descend until the approach profile conditions are met. If the approach mode is selected when the aircraft conditions are below the approach profile, the DAFCS commands the aircraft through the longitudinal trim actuator to decelerate at 1 knot/second while in the radar altitude hold mode until the approach conditions are met. If the approach mode is selected when the aircraft is above the approach profile, the DAFCS commands the aircraft through the collective trim actuator to descend at 360 feet/minute while the aircraft is more than 50 feet above the approach profile or at 120 feet/minute when the aircraft is less than 50 feet above the profile, using the radar altimeter until the approach profile conditions are met. When the approach profile conditions are met, the aircraft simultaneously decelerates at 1 knot/second and descends at 120 feet/minute. This profile is maintained until the aircraft attains 1 knot of Doppler groundspeed and comes to within 1 foot of the selected radar altitude. If the selected altitude is below 40 feet, the aircraft flies to 40 feet and zero longitudinal groundspeed and then descends to the selected altitude. When groundspeed equals 1 knot or less and the aircraft altitude is within 2 feet of the selected altitude, the hover coupler mode automatically engages and the aircraft accelerates to the selected longitudinal groundspeed. In very calm sea conditions where the Doppler return signal is unreliable and the Doppler goes into memory, a no-Doppler approach is possible. In this condition the pilot flies the cyclic control, and the AFCS controls the rate of descent. A summary of automatic approach malfunctions is contained in Figure 2-39.

WARNING

Certain AFCS fail advisories during an automatic approach will cancel the automatic approach function. If this occurs when on or above the approach profile, the aircraft will remain trimmed in a descent with no altitude hold engaged. Immediate pilot action will be required to avoid water impact (see Figure 2-39).

2.9.4.10 Hover Coupler

The hover coupler provides longitudinal and lateral groundspeed control and stabilization about the selected groundspeed, and automatic altitude retention. The longitudinal and lateral groundspeed and the altitude are selectable on the AFCS CONTROL panel. Longitudinal and lateral groundspeed and the altitude are selectable on the AFCS and can also be beeped ±10 knots with the cyclic TRIM switch about the groundspeed selected on the AFCS CONTROL panel. The hover coupler mode is engaged automatically at the termination of the automatic approach, or can be engaged manually when the aircraft is hovering with less than 5 knots longitudinal groundspeed by pressing the APPR/HVR button on the AFCS CONTROL panel with SAS 2, TRIM, and AUTO PLT engaged. After engagement, the aircraft accelerates to the longitudinal and lateral groundspeeds selected on the AFCS CONTROL panel. Radar altitude hold engages automatically when the aircraft altitude is within 2 feet of the altitude selected on the HVR ALT control knob. Pressing and releasing the cyclic TRIM REL will remove cyclic trim switch inputs, returning the aircraft to the LONG VEL and LAT VEL settings on the AFCS CONTROL panel. Because of the Doppler noise, short-term longitudinal and lateral groundspeed is obtained from integrated longitudinal and lateral inertial acceleration. Long-term correction is obtained from the Doppler sensor using a 7 second filter. See paragraph 2.9.4.5 for a description of the altitude hold feature of the hover coupler.

2.9.4.11 Automatic Depart

The departure can be initiated at any time in the approach or from the coupled hover. The aircraft will assume an approximately 2° nosedown attitude to commence the acceleration.

Note

- The transition from a 5° noseup attitude in the final phases of the approach to a 2° nosedown attitude for departure acceleration may appear excessive to the pilot, but is normal and is no cause for concern.
- If yaw rates are in excess of 2.5°/second, AFCS will not roll the wings level at 60 knots. This is caused by the heading hold feature being unable to engage. The departure will continue in a flat turn.

The AFCS will initially maintain the track of the helicopter over the ground existing at the time the departure mode is selected. The AFCS stores the roll attitude occurring at the time the trim release button was last depressed, prior to or in the automatic approach. If the stored roll attitude is less than 4° angle of bank, the AFCS will level the wings and maintain heading as the aircraft accelerates through 60 KIAS. If the stored roll attitude is 4° angle of bank or greater, the AFCS will roll the helicopter to that attitude when 60 KIAS is exceeded and continue the departure in a spiral. Whether heading hold or coordinated turn mode is active in the departure, the aircraft will climb and accelerate to 500 feet AGL and 100 KIAS if not interrupted by the pilot.

EVENTS	RAD ALT FAILURE	DOPPLER FAILURE
During approach	Aircraft will continue in a descent all the way to the water (if on or above profile with RDR ALT light off). Flashing AFCS caution light and CPLR/ ALT FAIL ADVISORY light. APPR disengages.	Approach will continue using airspeed and altitude. Pilot controls airspeed and wing attitude using beeper trim. Flashing AFCS caution light. HVR bars freeze. DOPP flag appears in .AI CPLR FAIL ADVISORY light.
In a coupled hover or below descent Profile (RDR ALT HOLD ENGAGED)	RDR ALT Hold switches to BAR ALT Hold. Automatic approach/coupled hover disengages. Altitude hold retained. Flashing AFCS caution light and CPLR/ ALT FAIL ADVISORY light.	Coupled hover disengages. Attitude hold RDR ALT hold retained. Flashing AFCS caution light. HVR bars freeze. DOPP flag appears in AI. CPLR FAIL ADVISORY light.
During departure	Aircraft will climb through 500 feet. No altitude hold will engage. Airspeed will accelerate to 100 knots. Flashing AFCS caution light and CPLR/ ALT FAIL ADVISORY light.	Aircraft will climb to 500 feet and RDR ALT hold will engage. Airspeed will increase to approximately 65–75 knots (dependent upon nose attitude when failure occurred). Flashing AFCS caution light. HVR bars freeze. DOPP flag appears in AI. CPLR FAIL ADVISORY light.

Notes:

Doppler degradation can be classified as follows:

1. Doppler power or transmitter fail — Bars center.
2. Doppler memory or receiver fail — Bars freeze.

Figure 2-39. Automatic Approach Malfunction Matrix

The automatic depart mode provides the capability to perform an automatic departure from a coupled hover or from an automatic approach to a cruise airspeed of 100 KIAS and altitude of 500 feet. If the coupled hover or the automatic approach feature has already been engaged, the automatic depart mode is engaged by depressing the DEPART HOV button on the cyclic grip illuminating the green DEPART light on the AFCS CONTROL panel. Depressing the DEPART HOV button a second time will disengage the automatic depart mode and radar hold, returning aircraft control to the pilot. Radar altitude hold may be retained by depressing the collective trim switch prior to the second DEPART HOV button depression. Upon engagement, the aircraft accelerates at 2 knots/second and climbs at 480 feet/minute. During the departure, the DAFCS computer monitors engine torque to ensure it does not exceed 116 percent. At 100 KIAS, the airspeed hold automatically engages, and at a radar altitude of 500 feet, the radar altitude hold automatically engages. Any alternate cruise airspeed or altitude condition less than 100 KIAS and 500 feet can be attained by depressing the cyclic trim release, collective trim release at the desired airspeed and altitude respectively. If either TRIM REL button is depressed and released, the hold mode (airspeed or altitude) associated with that control axis is engaged, and the aircraft continues to follow the depart profile for the other axis until the final cruise condition for that axis is met.

Automatic depart mode is the outer–loop function operating through the pitch, roll, and collective trim actuators. As in the automatic approach mode, above 60 KIAS, roll attitude is maintained and below 60 KIAS the DAFCS commands roll to eliminate lateral drift.

WARNING

- If the DPRT button is not depressed during a waveoff from an automatic approach or departure from a coupled hover, the automatic approach will reengage after the cyclic and collective trim switches are released, causing the aircraft to resume a descending profile.
- It is necessary to depress the DPRT button twice during a manual waveoff to prevent torque limiting.

2.9.4.12 Crew Hover

The crew hover feature provides the crewman with the capability to position the helicopter during hoist and rescue operations. The crewman controls the aircraft from the crew hover–trim control panel. The panel is illustrated in Figure 2-40. The crew hover controller has a control authority of ±5 knots laterally and longitudinally about the reference values selected on the AFCS CONTROL panel LONG VEL and LAT VEL control knobs plus the speeds beeped from the cyclic trim beep switch. The crew hover feature is activated from the AFCS CONTROL panel by depressing the CREW HVR button and can only be activated if the hover coupler mode is already engaged. If the automatic depart mode is activated while crew hover is engaged, crew hover will be disengaged and the automatic depart mode will be engaged.

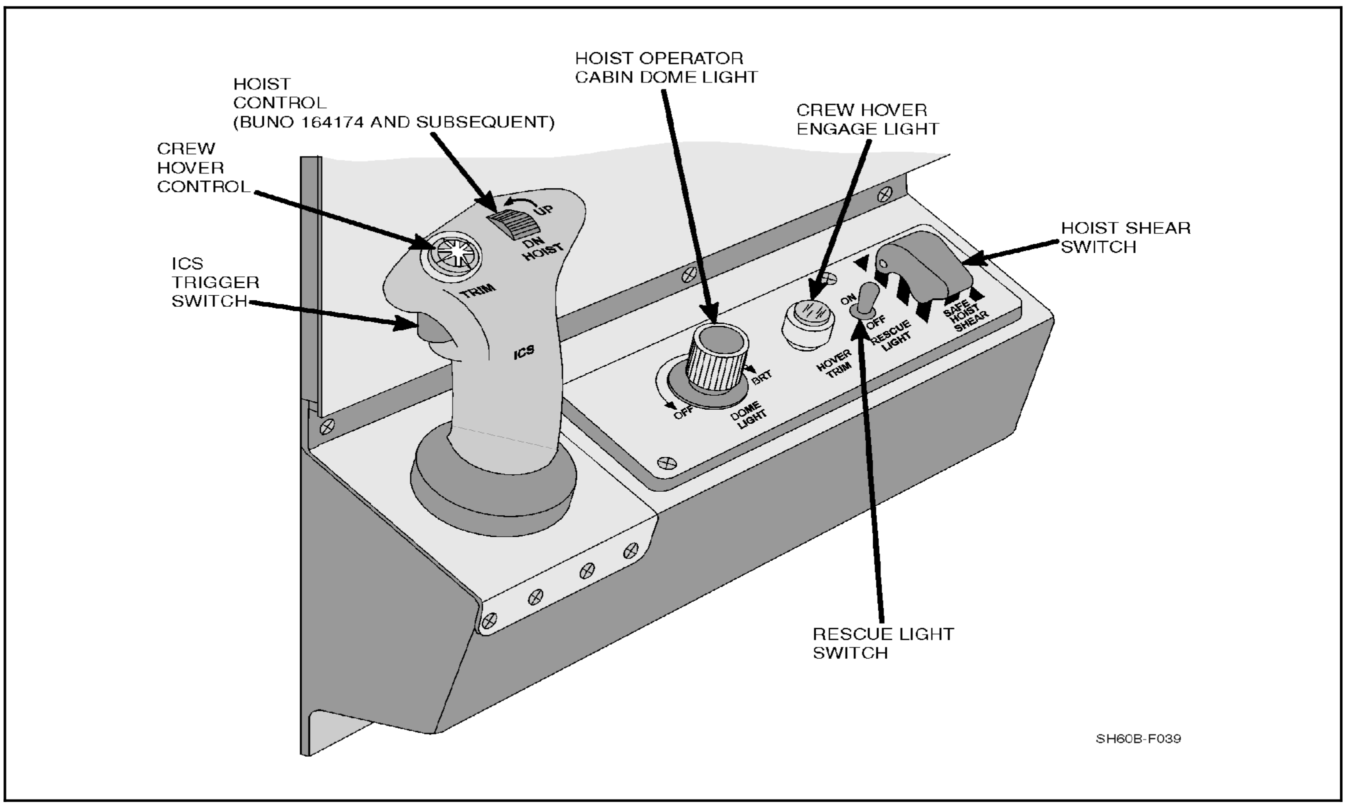


Figure 2-40. Crew Hover–Trim Control Panel

2.9.4.13 Pitch Bias Actuator

The Pitch Bias Actuator (PBA) provides longitudinal cyclic displacement proportional to airspeed. The DAFCS commands the PBA as a function of pitch attitude, pitch rate, and airspeed. The PBA is an electromechanical series actuator with ±15 percent control authority and ±3 percent per second rate limit. The PBA functions automatically upon application of power to the DAFCS computer and is not selectable on the AFCS CONTROL panel. The DAFCS computer monitors the PBA position to confirm correct response to the input commands. If the PBA fails, the DAFCS lights the BIAS advisory light on the AFCS CONTROL panel and flashes the AFCS DEGRADED light on the caution/advisory panel and commands the PBA to a predetermined position depending on the type of failure. PBA failure modes are:

1. Attitude failure: bias actuator centered.
2. Pitch rate failure: faded out pitch rate component.
3. Airspeed failure: actuator goes to 120 knot position and attitude and rate continues to function.
4. Actuator failure: power removed from actuator.

If the malfunction that caused the shutdown was of an intermittent nature, the actuator operation can be reset by pressing the appropriate MODE RESET button.

WARNING

When flying with the BIAS FAIL ADVISORY light on, up to 1 1/2 inches of forward or aft cyclic control authority may be lost.

2.9.4.14 Blade Fold System

Automatic blade folding (Figure 2-41) is accomplished by an electromechanical fold mechanism. Pitch locks fix the pitch of the rotor blades in order to maintain clearance during the fold sequence. Two blade lockpin pullers, lock and unlock the hinge when folding and spreading the blades. Microswitches provide input signals to the blade fold electrical control system, which sequences blade folding, and also provides light indications to the BLADE FOLD control panel (Figure 2-42).

2.9.4.14.1 Blade Folding

Control switches and indications are located on the BLADE FOLD control panel. With the BLADE FOLD MASTER switch ON, the main rotor head will turn to the indexed position after the BLADE FOLD switch is placed to FOLD and the rotor brake is released. The INDEXED status light and the ROTOR BRAKE APPLY light will illuminate upon completion of the indexing cycle. Engaging CMPTR PWR/RESET and TRIM, and applying the rotor brake will permit the blade fold sequence to continue. The DAFCS commands the trim actuators to position the flight controls and allows the main rotor head pitch locks to engage. The blades are then permitted to move to their folded positions.

2.9.4.15 Pylon Fold System

The tail pylon is manually folded and unfolded. PYLON FLIGHT and FOLDED lights, located on the BLADE FOLD control panel, indicate pylon fold status. There are five microswitches, which set the PYLON FLIGHT light: the pylon lockpin switch, 5° switch, tail rotor blade indexer switch, and two stabilator lockpin switches. With the BATT switch ON, the tail rotor will index and lock when the pylon has folded 5°.

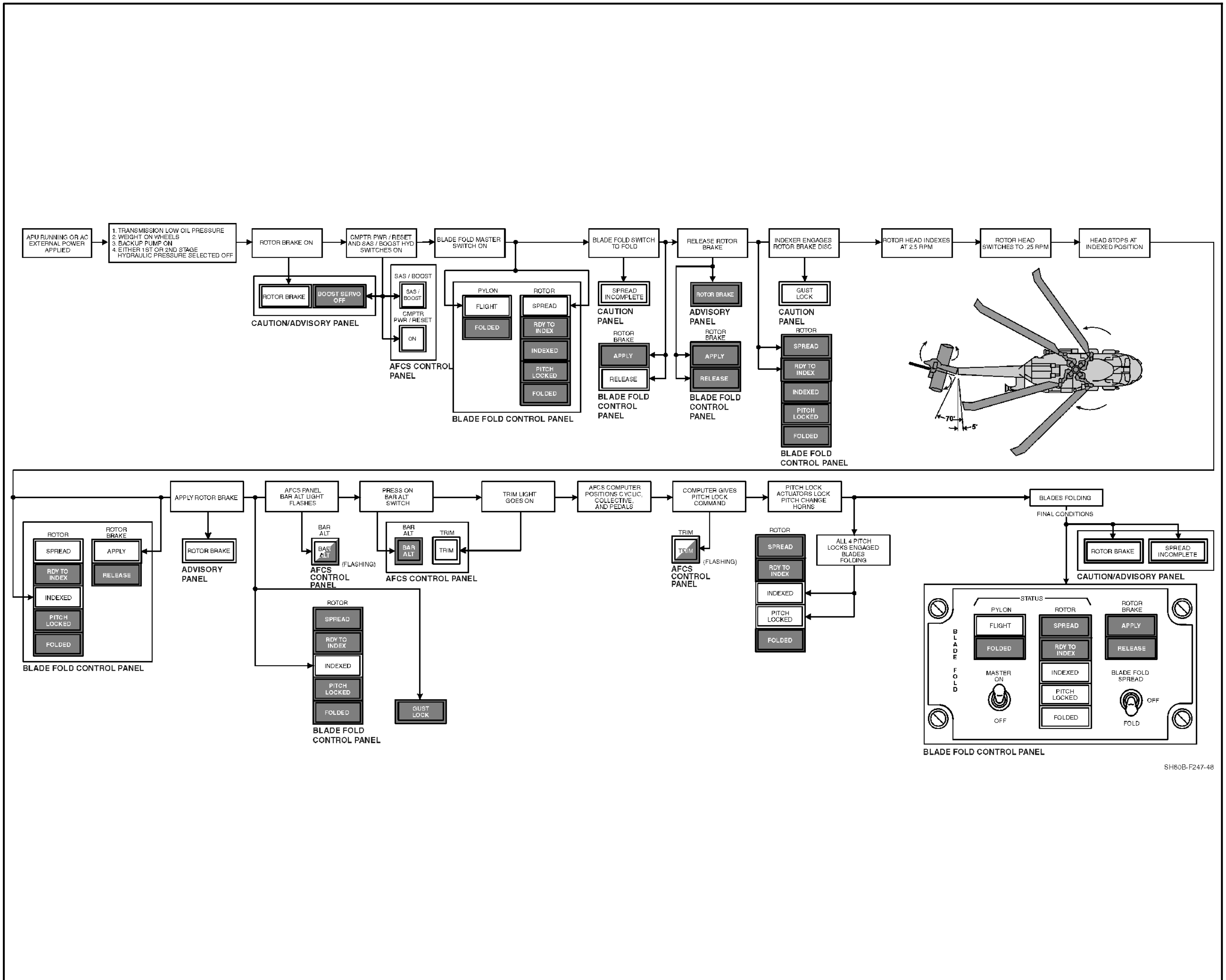


Figure 2-41. Blade Fold/Spread System (Fold Sequence) (Sheet 1 of 2)

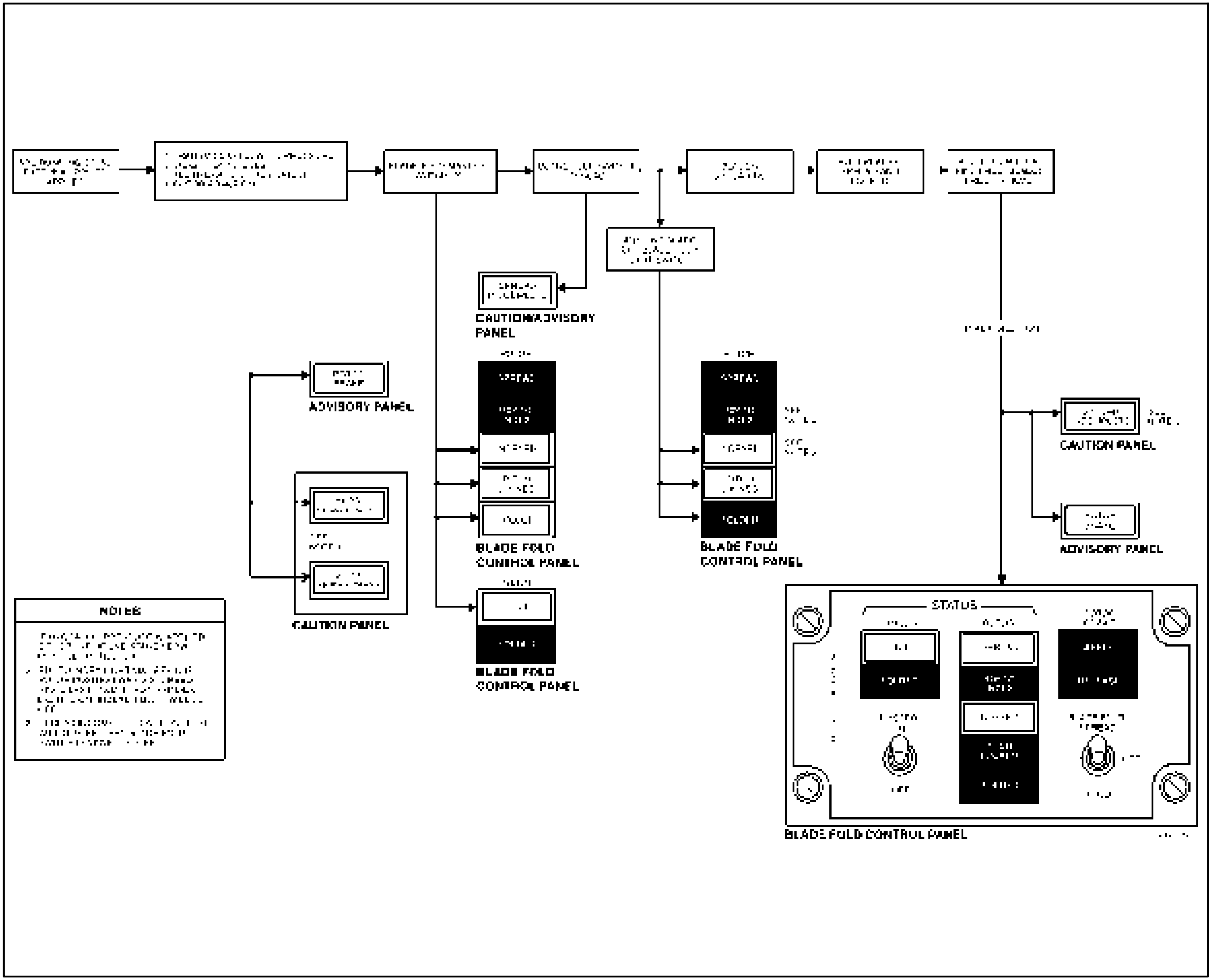
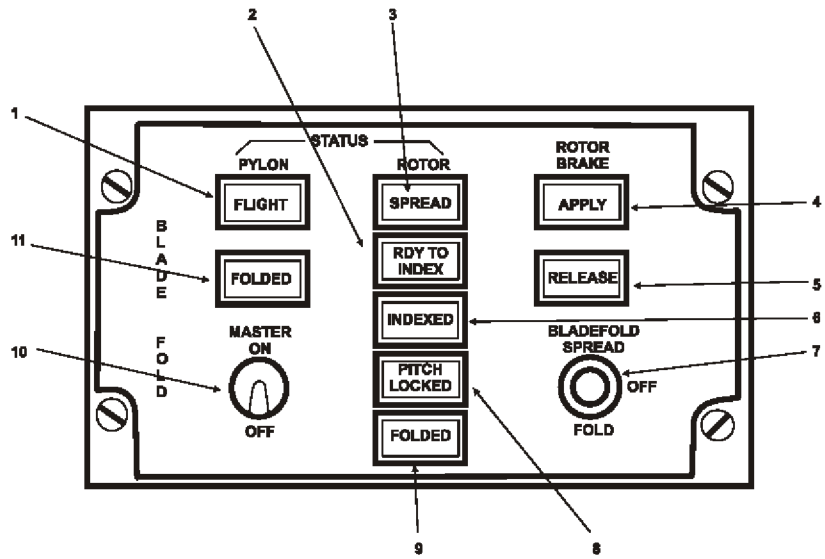


Figure 2-41 Blade Fold/Spread System (Spread Sequence) (Sheet 2)



INDEX	CONTROL	FUNCTIONAL DESCRIPTION
1	FLIGHT	All blade fold functions completed on tail pylon. Ready to Fly.
2	RDY TO INDEX	Main Rotor index pinion engaged in rotor brake disc. Index motor will drive rotor to index position when rotor brake is released.
3	SPREAD	Blade lockpins extended with main rotor index pinion retracted. Pitch locks are retracted.
4	APPLY	Commands rotor brake applied after rotor head is indexed. Blades are driven to the correct pitch for folding by engaging the CMPTR PWR/ RESET and TRIM pushbuttons.
5	RELEASE	Commands rotor brake released. When released, main rotor rotates to index position.
6	INDEXED	Rotor head indexed. Main rotor index pinion remains engaged in rotor brake disc until rotor brake is applied.
7	BLADE FOLD switch	Selects mode of operation (SPREAD, OFF, FOLD).
	SPREAD	Initiates spread sequence.
	FOLD	Initiates fold sequence.
8	PITCH LOCKED	Pitch locks engaged.
9	FOLDED	All four main rotor blades are folded.
10	BLADE FOLD MASTER switch	Applies or removes blade-fold power (OFF or ON).
11	FOLDED	Stabilator lock pins out, tail rotor indexed, pylon folded.

Figure 2-42. Blade Fold Controls

CAUTION

Unless external power is applied or the BATT switch is ON prior to folding the tail pylon, the tail rotor index actuator will not engage after starting the pylon fold sequence and uncontrolled tail rotor windmilling may result.

Note

Failure to suppress the DECU numerical fault codes on the PDU will prevent the automatic blade fold from operating due to the torque signal being relayed to the AFCS computer.

2.9.4.16 Automatic Preflight Checks

The DAFCS provides an automatic preflight check of the SAS components prior to flight. The automatic preflight is engaged using the SAS 1 switch with the following conditions:

1. Weight on wheels.
2. Rotor brake on.
3. Engine torques equal to zero.
4. CMPTR PWR/RESET engaged for at least 2 minutes (AFCS DEGRADED not illuminated).

Upon engagement, all rate gyros are automatically torqued to predetermined rates and checked for magnitude and polarity of rate gyro response. Simultaneously, the response of SAS 1 to rate gyro inputs is compared against a digital model. Failures are displayed on the fail status panel and stored in a BIT code display. CMPTR PWR/RESET switch has to be engaged at least 2 minutes to ensure gyros are up to speed. After gyros are up to speed, preflight requires approximately 10 seconds to complete.

2.10 LANDING GEAR

The landing gear (Figure 1-3) is a fixed main/tail, gear–type configuration. It consists of two single–wheel, main landing gears and a dual–wheel swivel–type tail gear. The long stroke of both main and tail wheel shock struts is designed to dissipate high–sink–speed landing energy without exceeding the ship deck strength limits. Wheel brakes are mounted on each main gear. Axle and high tiedowns are provided at each main gear, fuselage attachments are provided above the tail gear for tiedown, and connections to the shipboard tail–guide winch system are provided for RAST straightening and traversing.

WARNING

Flying with a stuck WOW switch will disable WOW functions including emergency jettison circuits, radar altimeter low altitude aural warning, Engine Out, and low rotor rpm lights. Pulling the WOW circuit breaker will not restore proper operation of some WOW functions in the air. Pulling the WOW circuit breaker in flight may disable the LOW ROTOR RPM light and the #1 and #2 ENG OUT warning lights.

2.10.1 Main Landing Gear

Each single–wheel main gear is mounted on a drag strut that trails aft from a pivot point mounted on the fuselage. A separated air/oil–type shock strut is mounted on the fuselage and to the aft end of the drag strut. A landing gear

WOW switch is installed on the left landing gear to prevent or control operation of certain systems when the weight of the helicopter is on the landing gear. The equipment that uses the WOW switch is shown in Figure 2-44.

2.10.1.1 Wheel Brake System

Main landing gear wheels have self-contained, self-adjusting disc hydraulic brakes (Figure 2-45). The wheel brake system consists of four master cylinder/reservoirs, two slave valves, a parking brake valve, and two wheel brake assemblies. A master cylinder is connected to each rudder pedal. The purpose of the slave valves is to give the pilot or ATO the ability to apply the brakes. Each wheel brake consists of a steel rotating disc, brake pucks, and a housing that contains the hydraulic pistons. The brakes have a visual brake-puck wear indicator. The parking brake handle, marked PARKING BRAKE, is located on the right side of the center console and allows the brakes to be locked by either the pilot or ATO after brake pressure is applied. The parking brakes are applied by pressing the toe brake pedals, pulling the parking brake handle up to its fully extended position, and then releasing the toe brakes while holding the handle up. The PARKING BRAKE ON advisory light will illuminate. The advisory light only indicates that the parking brake handle is up. Pressing either the pilot or ATO left brake pedal will release the parking brakes, the handle will return to the off position, and the advisory light will extinguish. Power is provided to this advisory system by the NO. 1 DC primary bus through a circuit breaker marked EXT ADVSY LTS and located on the SO circuit breaker panel.

2.10.2 Tail Landing Gear

The tail landing gear is of a cantilevered design, with an integral shock strut capable of swiveling 360°. The two tail gear wheels are mounted on a splined axle incorporated in the shimmy damper (Figure 2-43). The shimmy damper causes both of the tail gear wheels to rotate at the same rate, preventing aircraft tail oscillations during taxi, takeoff, and running landings. For helicopter guidance for traversing on the flight deck, a RAST tail probe and probe actuator are mounted on the tail gear.

2.10.2.1 Tail Wheel Lock

The tail wheel lock is extended and retracted by an electric motor-operated actuator located on the tail wheel shock strut housing. The tail wheel lock switch is located on the forward side of the parking brake handle in the cockpit. Unlocking the tail wheel illuminates the TAIL WHEEL UNLOCKED advisory light on the caution/advisory panel and unlocks the tail wheel lock pin. Power to operate the tail wheel lock motor is provided by the DC essential bus through a circuit breaker marked TAIL WHEEL LOCK on the overhead console. A manual lock release is located on the strut. In the down position, the tail wheel is manually unlocked. It cannot be controlled electrically. In the up position, the tail wheel lock is controlled by the switch on the parking brake handle. There is no manual locking feature.

2.10.2.2 Tail Bumper

A nitrogen-filled tail bumper (Figure 1-3), mounted on the underside of the tail pylon, prevents the stabilator from striking the ground when landing with nose-high attitudes.

2.11 RAST SYSTEM

The air vehicle portion of the RAST system (Figure 2-46) comprises a control panel, a main probe assembly, and a tail probe assembly. RAST is capable of:

1. Assisting the pilot in landing the helicopter on the flight deck.
2. Securing the helicopter to the deck.
3. Straightening the helicopter to a laterally centered position.
4. Traversing the helicopter into and out of the hangar.

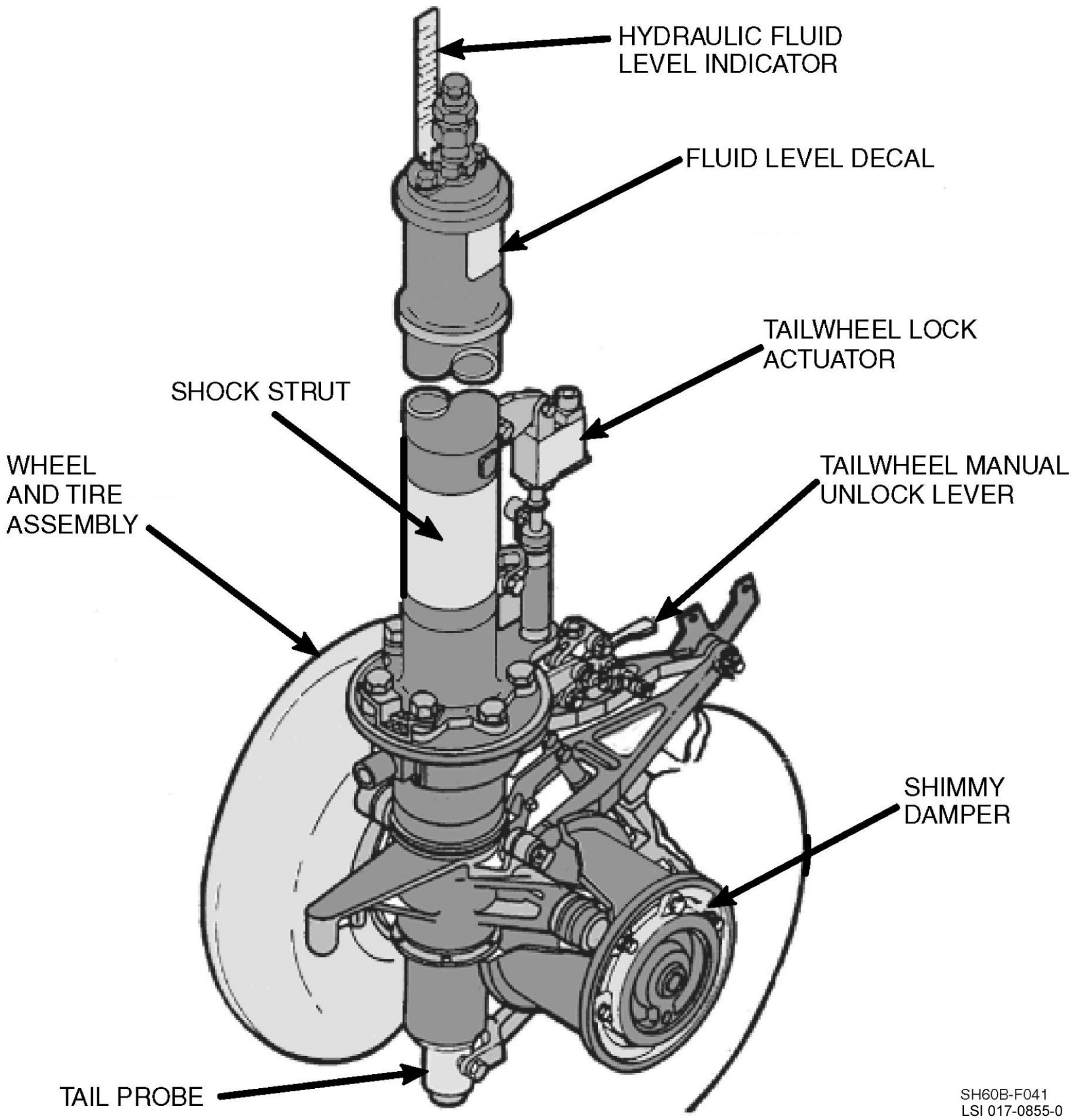


Figure 2-43. Tail Wheel Assembly

INTERFACE	FUNCTION AFFECTED	FLIGHT	GROUND
Blade fold	Blade fold actuators	Disabled	Enabled
AFCS	Pitch hold	Enabled	Disabled
	Heading hold	Enabled	Disabled
	Automatic preflight checks	Disabled	Enabled
Generator control unit	Underfrequency protection	Disabled	Enabled
Master shear	RAST shear	Enabled	Disabled
	Cargo hook emergency release	Enabled	Disabled
	MAD shear	Enabled	Disabled
	Rescue hoist	Enabled	Disabled
Sonobuoy launcher	Rotary valve to DECU circuitry	Enabled	Disabled
Hydraulics	U-1, U-2 logic modules	Enabled	Enabled with BACKUP HYD PMP-AUTO
			Disabled with BACKUP HYD PMP-OFF
	LDI test	Disabled	Enabled
Master caution panel	Engine out lights	Enabled	Disabled
	Low rotor rpm lights	Enabled	Disabled
Tail rotor blade positioner	Tail indexing	Disabled	Enabled
Radar altimeter	Height indicator low altitude aural warning	Enabled	Disabled
Armament	Master armament and jettison circuits	Enabled	Disabled
FLIR turret	LRD	Enabled	Disabled
M299 launcher	M299 launcher arming	Enabled	Disabled
SO console	Search radar (cannot be bypassed)	Enabled	Disabled
Mission avionics rack	KIT and KIR zeroize logic		
	Mechanical hold	Disabled	Enabled
	Electrical hold	Enabled	Disabled
	Data-link transmitter	Enabled	Disabled

Figure 2-44. Weight–On–Wheels Functions

2.11.1 RAST Control Panel Indicators

The RAST control panel is located on the SO console above the RADAR/DISPLAY control panel and provides control and status indication of the system (Figure 2-47). The cyclic grips have an electrical release button to free the recovery assist (RA) cable from the aircraft. An emergency release (EMER REL) button to shear the main probe messenger cable is located on the cyclic grip. A mechanical emergency release handle, located on the left side of the center console, is used to release the RA cable from the probe in the event of electrical release actuator failure.

The MASTER switch is a two–position switch and, when placed in the ON position, supplies power to the control portion of the system. The MAIN PROBE, UP/DOWN switch is a three–position switch, spring–loaded to the center. The MESSGR CABLE–UP/DOWN switch is spring loaded to the center position. When the switch is placed in the UP position, the messenger cable is reeled into the aircraft, and when it is placed in the DOWN position, the cable will reel out from the aircraft. The main probe must be extended before the messenger cable can be reeled out, and the messenger cable must be seated in order to retract the main probe.

Note

The DOWN light is not a positive down indication for the tail and main RAST probe (electrically actuated only).

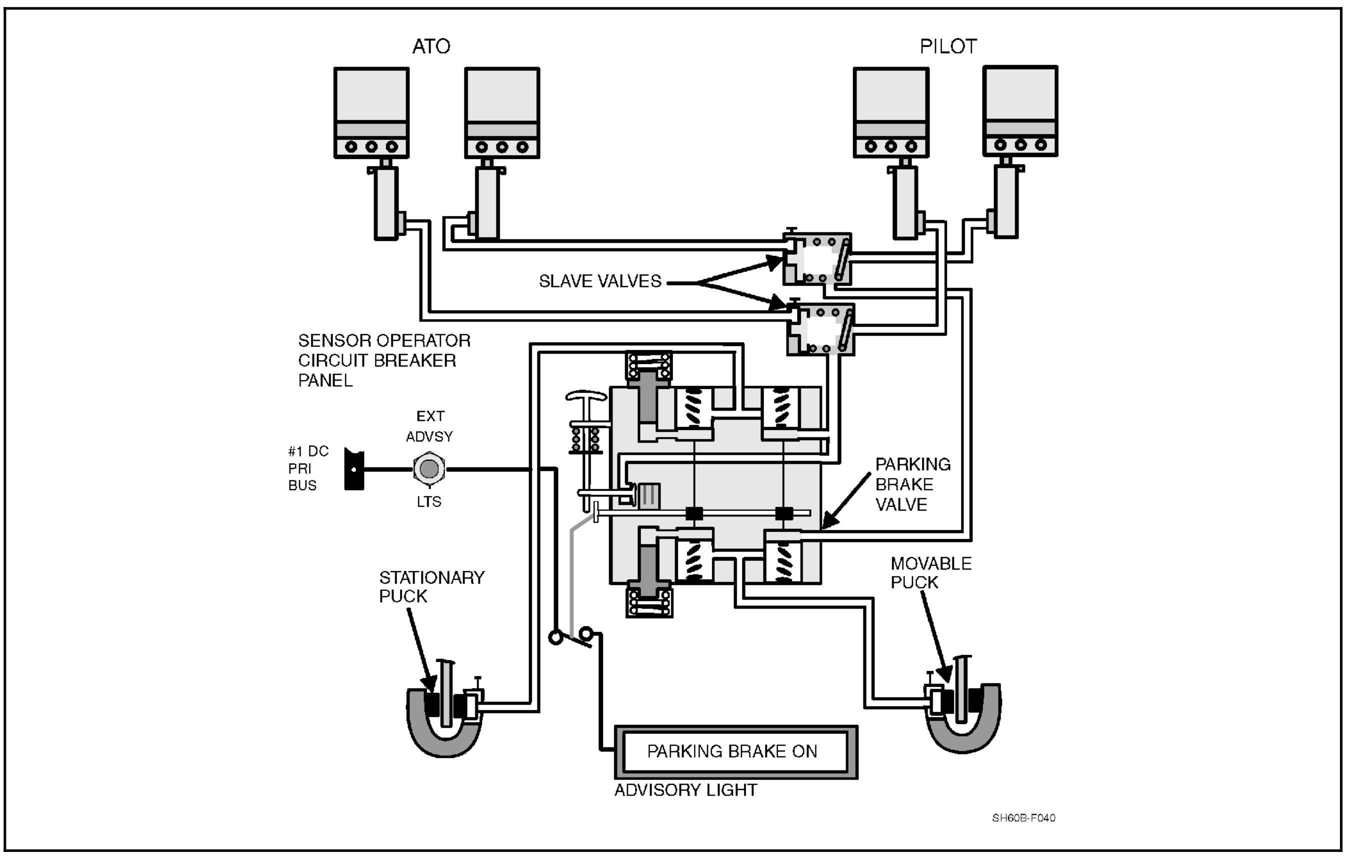


Figure 2-45. Wheel Brake Schematic

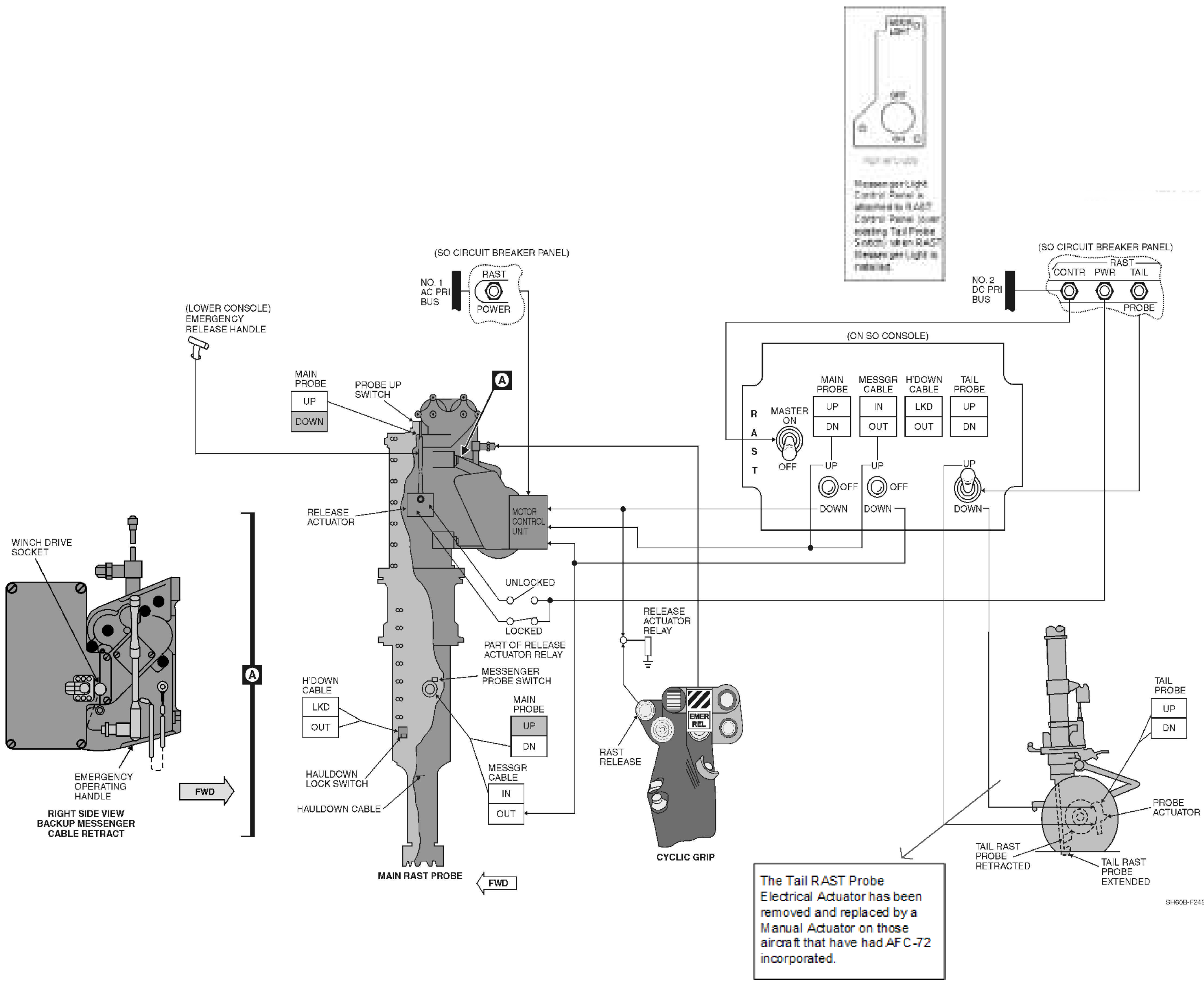


Figure 2-46. RAST Block Diagram

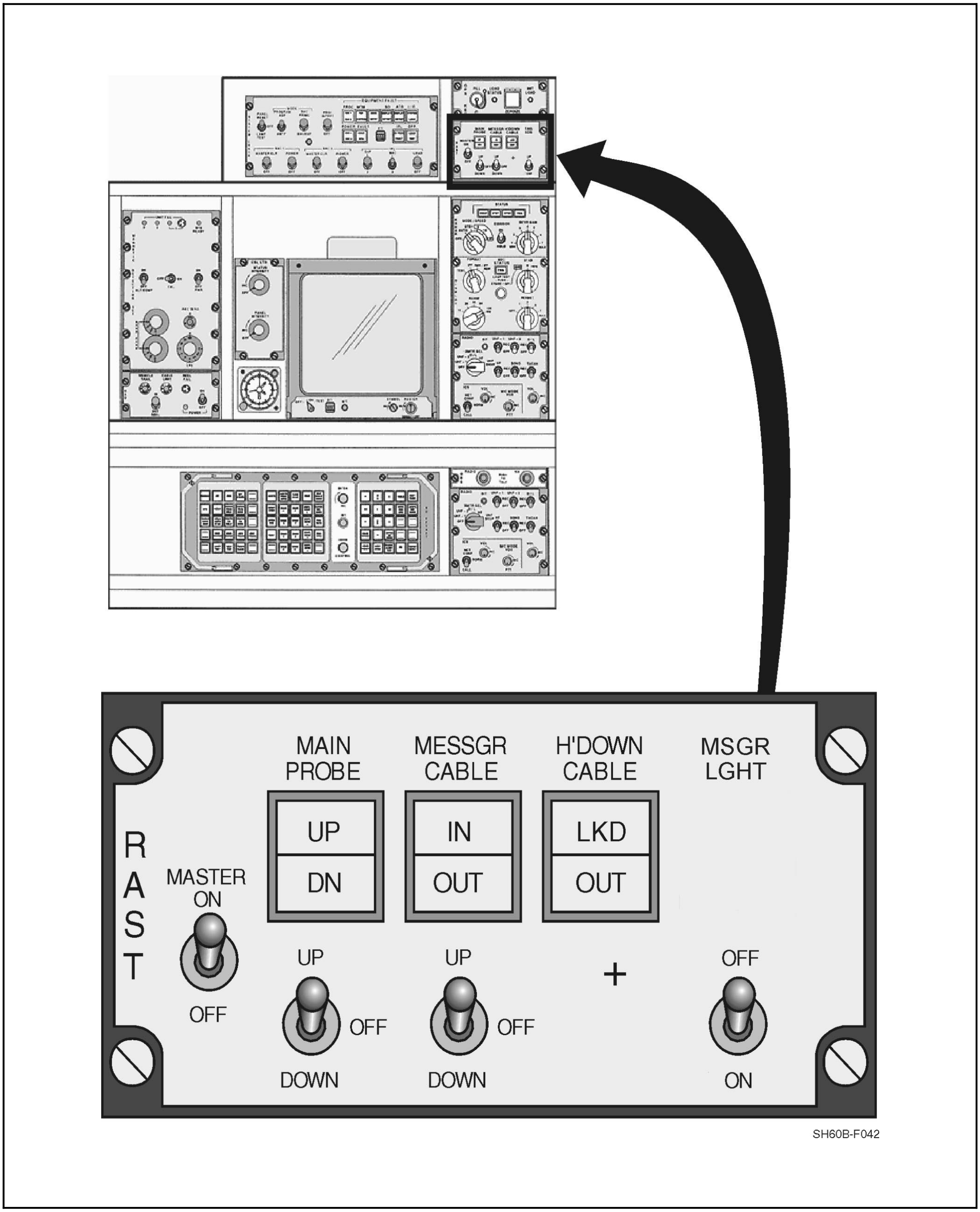


Figure 2-47. RAST Control Panel

The MESSGR CABLE light gives the status of the cable. When it displays IN, the messenger cable is retracted fully into the main probe. When OUT is displayed, the messenger cable is extended. The H'DOWN light gives the status of the RA cable. A LKD indication is displayed when the RA cable from the surface ship is locked into the main probe. The OUT indication is displayed when the RA cable is free of the main probe. The main probe (Figure 2-46) contains a 3 phase, 115–Vac, 400–Hz electrically operated hoist, which lowers the probe, reels out the messenger cable, raises the RA cable, and retracts the probe. The hoist is powered by the NO. 2 AC primary bus through a circuit breaker marked RAST POWER. This circuit breaker is located on the SO circuit breaker panel. An electrically fired guillotine is provided to shear the messenger cable in an emergency. The guillotine is powered by the NO. 2 DC primary bus through a circuit breaker on the ATO circuit breaker panel marked RAST SHEAR. Attached to the probe is a release actuator, used to release the RA cable from the locks in the probe. It is also used to lock the probe in the retracted position. Three switches are mounted on the probe: the probe UP switch, the messenger probe switch, and the H'DOWN LKD switch.

2.11.2 Main RAST Probe

The airborne provision for the RA system is a fully retractable main probe (Figure 2-46). It is mounted on the centerline of the aircraft near the center of gravity. In a fully retracted position, it is held in an uplock. From there, it is spring loaded to a fully extended position for landings into the rapid securing device (RSD). The probe has an electrically powered hoist mounted to its outer housing. The hoist provides the messenger cable, deployed through the center of the probe, to retrieve the surface ship RA cable. The messenger cable end fitting provides a snap–in connection for the RA cable. After pulling the RA cable into the locked position, the messenger cable is automatically disconnected from the RA cable. The lower end of the extended probe is designed to be captured by the RSD after landing. With the probe secured within the RSD, the helicopter is held against horizontal and/or axial tension loads. A swiveling crenelated ring on the end of the probe is provided for axial loads. The main probe is powered by the NO. 2 DC primary bus through a circuit breaker on the SO circuit breaker panel marked RAST PWR.

2.12 FLIGHT INSTRUMENTS

The electrically operated instruments function on alternating current, direct current, or both and are protected by appropriately marked circuit breakers on the pilot and ATO circuit breaker panels.

2.12.1 Pitot Static System

The pitot–static system provides pressure for the operation of the differential pressure instruments (barometric altimeters and airspeed indicators). Two pitot tubes are mounted on the nose, forward of the cockpit. Two static ports are located on the fuselage sides, aft of each cockpit door (Figure 1-3). Each pitot head assembly consists of a baseplate with a strut and probe tube and an electrical connector, wired to two deicing heaters in the tube. Pitot pressure is sensed at the opening of the forward end of each tube. Static 1 and static 2 pressures are sensed through ports aft of the cockpit doors. Pitot pressure is supplied from the pitot tubes to the airspeed indicators, airspeed and air data transducers, and pitot–drain caps. To obtain a difference in the pressure for operation of the barometric differential pressure instruments, static air pressure (atmosphere) is supplied through the static ports to the altimeters, airspeed indicators, airspeed and air data transducers, and static drain caps.

2.12.1.1 Airspeed Indicator

Two airspeed indicators are installed on the instrument panel (Figure 1-8) for the pilot and the ATO. The indicators are differential pressure instruments, measuring the difference between impact pressure and static pressure. System error is noted on placards located below the instrument panel on each side of the lower console (Figure 1-7).

2.12.1.2 Barometric Altimeter/Encoder

The AAU–32/A Altimeter/Encoder functions as a barometric altimeter for the pilot and a pressure altitude sensor for the AN/APX–100 IFF Transponder. The altimeter/encoder is on the pilot side of the instrument panel (Figure 1-8). The operating range of the altimeter is from –1,000 to +50,000 feet. The barometric pressure–set knob permits altimeter settings from 28.10 to 31.00 inches Hg. A window in the lower right section of the altimeter displays the selected altimeter setting. The altimeter is equipped with a continuously operating DC–powered vibrator to improve altitude indicating accuracy. The encoder provides a digital output of pressure altitude in units of 100 feet to the IFF transponder, with mode C selected, for automatic pressure altitude transmission. The encoder operates throughout the operating range of the altimeter, but, unlike the altimeter, it reports altitude using a permanent altimeter setting of 29.92 inches Hg. If there is a loss of 115 Vac, 400 Hz power, the warning flag on the pilot altimeter indicator, marked CODE OFF, will be displayed. The NO. 2 DC primary bus furnishes power to the pilot altimeter through a circuit breaker on the ATO circuit breaker panel, marked PILOT ALTM. The NO. 1 DC primary bus furnishes power to the ATO altimeter through a circuit breaker marked ATO ALTM.

2.12.1.3 Barometric Altimeter

The AAU–31/A Altimeter, installed on the left side of the instrument panel (Figure 1-8), is identical to and operates in the same manner as the pilot AAU–32/A Altimeter/Encoder, except that there is no encoder associated with the altimeter, and there is no warning flag on the indicator.

2.12.1.4 Vertical Speed Indicators

Two indicators on the instrument panel (Figure 1-8) indicate vertical speed in thousands of feet per minute. The first 1,000 feet are marked in 100–foot gradations. Each vertical speed indicator (VSI) independently reads static cabin pressure through a port in the back of each gauge.

Note

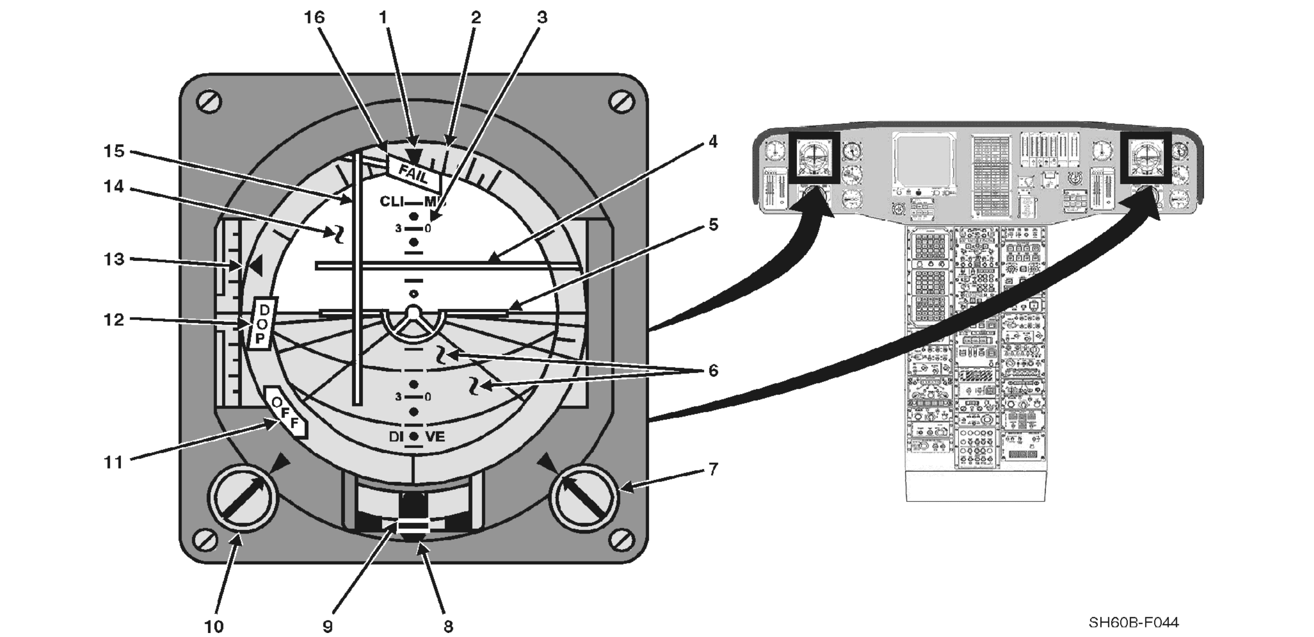
- Vertical speed indicators are unreliable during transition to ground effect.
- The VSI may momentarily indicate a false rate of descent while opening the cargo hatch cover in flight.

2.12.1.5 Attitude Indicator

Identical Attitude Indicator (AIs) are located on the pilot and ATO instrument panel. These indicators furnish a visual display of aircraft attitude. Figure 2-48 shows the AI and describes the individual indicating elements. Power for the pilot AI is supplied from the AC essential bus and for the ATO from the NO. 1 AC primary bus, both through the center circuit breaker panel, marked PILOT AI and ATO AI, respectively.

2.12.2 Radar Altimeter, AN/APN-194

The radar altimeter (RAD ALT) (Figure 2-49) is a range–tracking radar that provides continuous measurement of height above land or water. It has a range of 0 feet to 5,000 feet, with an accuracy of ±3 feet or ±4 percent, whichever is greater; however, only a range of 0 feet to 1,000 feet is indicated on the instrument. Tracking above 1,000 feet is used by the operational navigation system and the AFCS. On deck, a reading of 0 to 7 on the navigation table is permissible. Moving either pilot or ATO height indicator control knob out of the OFF position will provide height indication to both radar altimeters.



INDEX NUMBER	CONTROL	FUNCTIONAL DESCRIPTION
1	Bank pointer	Indicates bank or roll angle.
2	Bank scale	Measure of bank angle. Scale marks indicate 5° each.
3	Pitch reference scale	Measure of pitch angle. Each interval between dot and line is 5°.
4	VHA pointer	Indicates velocity along heading axis. One scale marking = 5 kt (±25 kt full scale).
5	Aircraft reference	Fixed miniature aircraft for attitude orientation reference.
6	Ground perspective line	Lines that show perspective of a grid of imaginary lines on the surface of the Earth.
7	Pitch trim knob	Adjusts attitude sphere for pitch trim.
8	Turn rate indicator	Indicates rate of turn. A standard rate turn is one needle width with ECP3032 installed and two needle widths without.
9	Slip/skid indicator	Indicates direction of slip or skid.
10	Roll trim knob	Adjusts attitude sphere for roll trim.
11	OFF flag	Indicates absence of internal power or absence of external ground signal. If received in flight, select alternate AGCA on mode select panel.
12	Doppler warning flag	Indicates Doppler radar is off or Doppler data not dependable.
13	VZA pointer	Indicates velocity along vertical axis. Each scale marking = 100 fpm (±500 fpm full scale).
14	Attitude sphere	Sphere that moves in two rotational degrees of freedom to indicate attitude of aircraft in bank and pitch.
15	VDA pointer	Drift velocity pointer. Indicates velocity across track. One scale marking = 5 kt (±25 kt full scale).
16	FAIL flag	Indicates failure of one or more internal status monitoring tests.

Note

Fast erection for AI is obtained through the ERECT pushbutton on the compass system controller, located on the lower console. Due to Doppler sensitivities to pitch, VZA provides only coarse rate information and should not be used as a precision hover reference.

Figure 2-48. Attitude Indicator (AI)

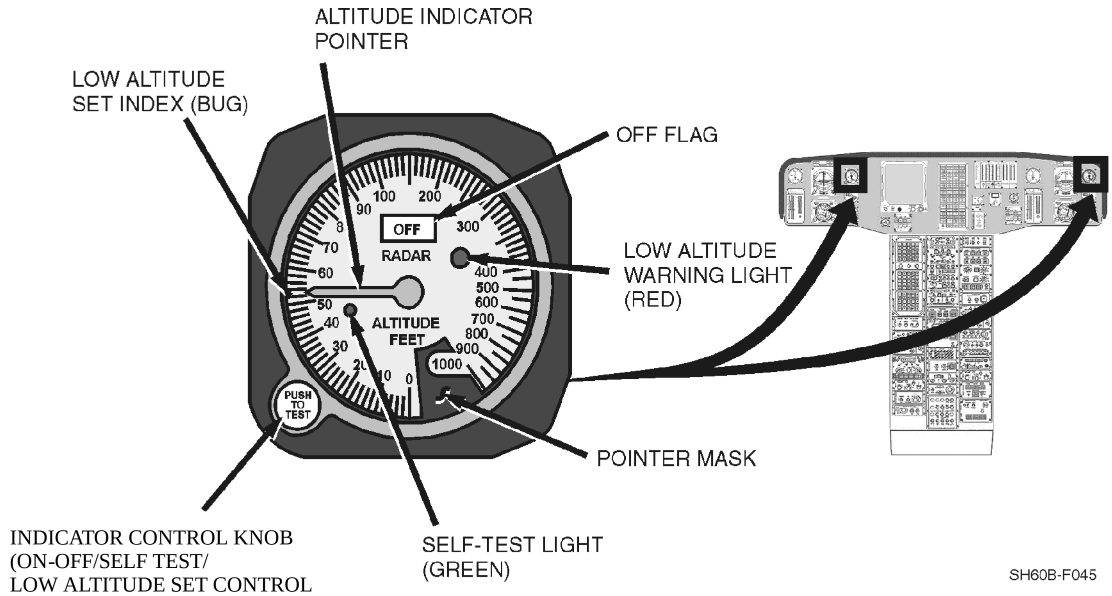


Figure 2-49. Radar Altimeter (Height Indicator)

The radar altimeter operates in two modes, search and track. In the search mode, the entire altitude range is searched for a ground return. In the track mode, the set locks onto the ground return and gives continuous altitude information to the height indicators. In addition, altitude data is sent to the data handling subsystem for processing. Each of the height indicators contains a complete radar altitude warning system (RAWS) function. The RAWS function provides a visual warning and an aural tone to the internal communications network. Even with RAD ALT switch off, pressing the test switch will provide a continuous beep in the headset.

Note

Pressing the RAD ALT TEST button above 5,000 feet activates a continuous beeping tone in both the pilot and ATO headsets. The tone can only be deactivated by turning off the RAD ALTs or descending below 5,000 feet AGL.

The RAWS feature of the AN/APN–194 is active when:

1. Fixed high altitude warning. When descending through 250 feet, the pilots will hear six beeps. This is disabled only if coupler is engaged.
2. Fixed low altitude warning (35 feet). When below the setting, the pilots will hear a continuous series of beeps. This is enabled only if coupler is engaged.
3. Variable altitude adjust. Each pilot will hear six beeps and the low altitude light will illuminate if the aircraft descends below his respective variable index.
4. Above 5,000 feet. If return is unreliable, height indicator will display OFF flag.
5. Power source failure to RAD ALT will cause height indicator to display OFF flag with continuous beeping tone.

The warning consists of a light on the indicator and a tone in the pilot and ATO headsets. The tone is a nominal 1,000–Hz signal, pulsed at a 2 cycle per second rate. Both stations will be alerted for the preset high and low altitude indices.

The electronic altimeter set consists of a receiver/transmitter, height indicators (pilot and ATO) (two), and antennas (left and right) (two). Power is supplied from the AC essential bus through the center circuit breaker panel marked RDR ALTM R/T. The pilot and ATO height indicators are powered by the DC essential bus through the overhead circuit breaker panel, marked HEIGHT IND PILOT and HEIGHT IND ATO, respectively.

Note

The bearing-distance-heading indicators (BDHI) and the compass system are discussed in Chapter 16.

2.12.3 Miscellaneous Flight Instruments

2.12.3.1 Standby Magnetic Compass

A lighted magnetic compass is installed above the instrument panel on the right-center windshield frame (Figure 1-5). The compass is used as a standby instrument for heading references. A compass correction card, with deviation errors, is installed on the forward right of the overhead console (Figure 1-6).

2.12.3.2 Outside Air-Temperature (OAT) Indicator

An ambient air-temperature indicator (Figure 1-5), marked FREE AIR, extends through the upper-center windshield panel. The direct reading instrument is marked in degrees Celsius.

2.12.3.3 Clock

Two 8-day clocks are installed on the instrument panel (Figure 1-8). The elapsed time knob is on the upper-right corner of the clock. The clock is wound and set with a knob on the lower-left corner. The SO is also provided with a clock.

2.13 WARNING, CAUTION, AND ADVISORIES

2.13.1 Master Warning System

Two amber, master caution warning lights (Figure 1-8), for the pilot and ATO, marked MASTER CAUTION PRESS TO RESET, are located on the master warning panel. They light whenever a caution light lights. These lights alert the pilot and direct attention to the caution lights on the caution/advisory panel. During caution/advisory panel test, when the switch is released from the test position, the master caution lights will flash 16 times to indicate that chip caution panel circuits are going through self-test. An existing malfunction within those circuits will prevent flashing of the master caution lights. The master caution warning lights should be reset at once to provide a similar indication if a second condition or malfunction occurs while the first condition is present. The master caution warning light can be reset from either pilot position. Power for both of the master caution warning lights is provided from the DC essential bus through a circuit breaker marked CAUTN ADVSY PNL and located on the overhead circuit breaker panel.

Four red warning lights, also located on the master warning panel, require immediate action if they light. The markings are #l ENG OUT, #2 ENG OUT, FIRE, and LOW ROTOR RPM. The LOW ROTOR RPM warning light will flash at a rate of three to five flashes per second if rotor RPM drops below 96 percent. The ENG OUT warning lights will light at 55 percent N_g speed and below. Refer to Figure 2-50 for a brief description of each fault.

Note

The ENG OUT and the LOW ROTOR RPM warning lights are disabled with weight on wheels, but may still be tested with the caution/advisory test switch.

LEGEND	LIGHTING PARAMETER OR FAULT
#1 ENG OUT	Indicates that the NO. 1 engine N_g speed is ≤55 percent.
FIRE	Indicates that a fire detector has actuated a fire-warning circuit.
MASTER CAUTION PRESS TO RESET	Indicates that a caution light on the caution panel has been actuated by a failed system.
#2 ENG OUT	Indicates that the NO. 2 engine N_g speed is ≤55 percent.
LOW ROTOR RPM	Indicates that the rotor speed is ≤96 percent N_r.

Figure 2-50. Master Warning Panel

2.13.2 Caution/Advisory Light System

The caution/advisory panel (Figure 1-8) is located on the instrument panel. The caution section (the upper two-thirds) of the panel indicates certain malfunctions or unsafe conditions with amber lights. The advisory section (the lower one-third) of the panel shows certain noncritical conditions with green lights. Each light has its own operating circuit and will remain lighted as long as the condition that caused it to light exists. The caution/advisory panel contains a self-test system for all engine and transmission drivetrain chip detector lights. During caution/advisory panel test when the switch is released from TEST position, the self-test is activated. If a malfunction exists in one of the engine and transmission drivetrain chip detect lights within the caution panel, that respective light will flash. The caution and advisory lights are powered by the DC essential bus through a circuit breaker marked CAUTN ADVSY PNL on the overhead panel. (Refer to the major systems for a complete description of the caution/advisory panel lights. (Refer to Chapter 12 for a description of the caution/advisory legend.)

2.14 FIRE-DETECTION SYSTEM

The fire-detection system (Figure 2-51) provides a visual cockpit indication when infrared radiation, caused by a fire or extreme overheating, is detected in either engine compartment or the APU compartment. The system consists of three control amplifiers located in the left-hand junction box; five sensors (two in each engine compartment and one in the APU compartment); # 1 and # 2 ENG emergency off T-handle fire-warning lights located on the engine control quadrant; APU FIRE EXT T-handle fire-warning light and FIRE DET TEST switch located on the overhead console; and two FIRE warning lights on the pilot and ATO master warning panels.

When one of the sensors detects infrared radiation (fire), and no blue light (sunlight), it sends out a voltage to its associated control amplifier. Sunlight filtered through smoke or haze, or at sunrise or sunset, may trigger the flame detectors and cause a false fire indication. The control amplifier then provides a voltage to both master warning-panel FIRE lights and the proper T-handle lights. The FIRE DET TEST switch (Figure 1-6) on the overhead console is a three-position rotary switch used to check all components of the fire detection system except the flame detector (which must be tested with red light). In the OPER position, the fire sensors are connected up to their respective indicators. The NO. 1 test position checks the continuity of the wiring, amplifiers and monitoring lights for the firewall-mounted detectors, NO. 1 and NO. 2 engines, and the APU compartments. If operating properly, the master FIRE warning light, both ENG EMER OFF T-handles, and the APU FIRE EXT T-handle will illuminate. The NO. 2 test position checks the continuity of wiring, amplifiers, and monitoring lights for the NO. 1 and NO. 2 engine deck-mounted sensors. If operating properly, the master FIRE warning light and both ENG EMER OFF T-handles will illuminate. The APU FIRE EXT T-handle will be off. Electrical power for the engine compartment detectors is supplied by the DC essential bus through the FIRE DET NO. 1 and NO. 2 ENG circuit breakers on the overhead circuit breaker panel. The detector in the APU compartment is supplied by the battery bus through the APU FIRE DETR circuit breaker on the center console circuit breaker panel.

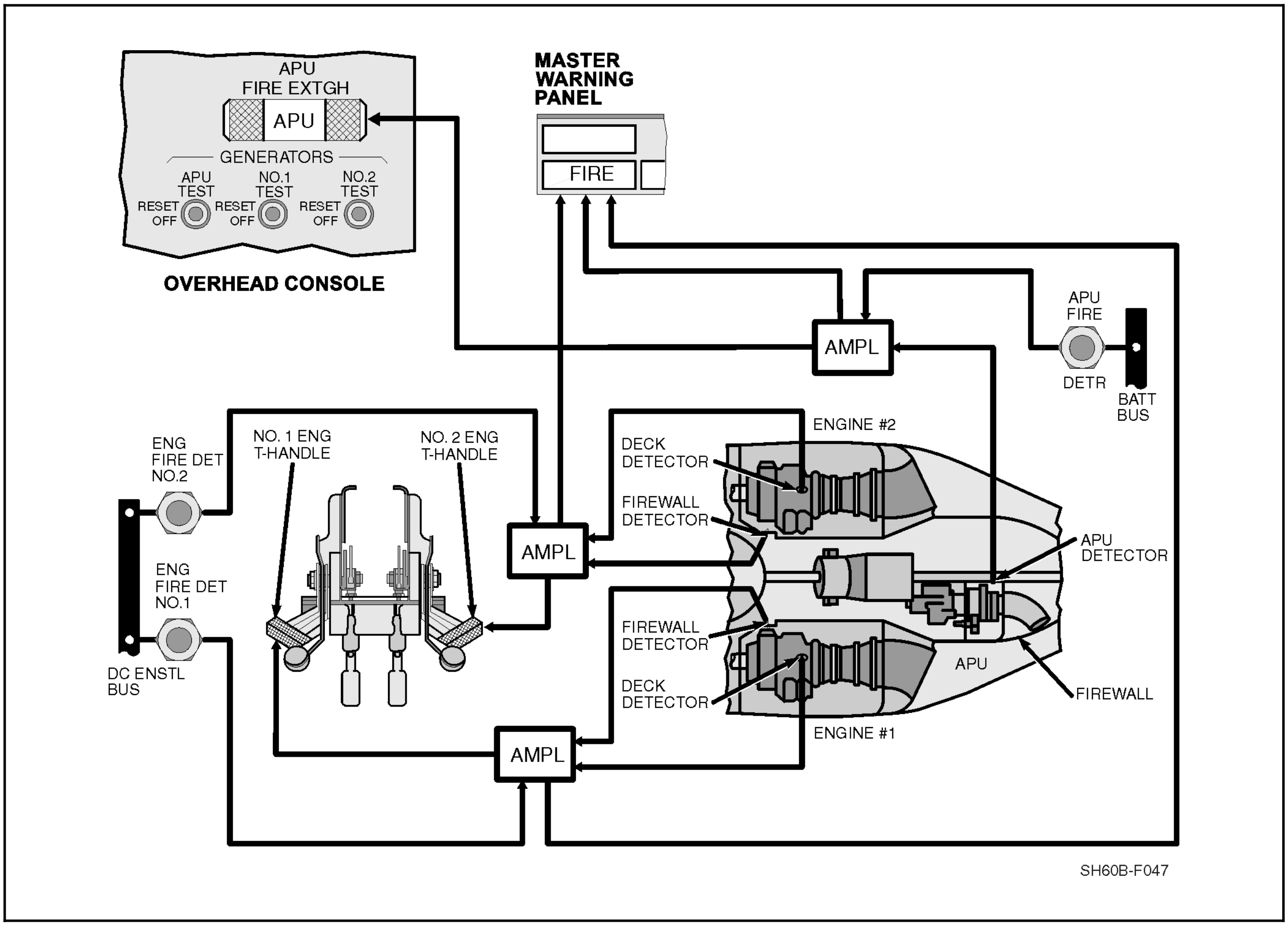
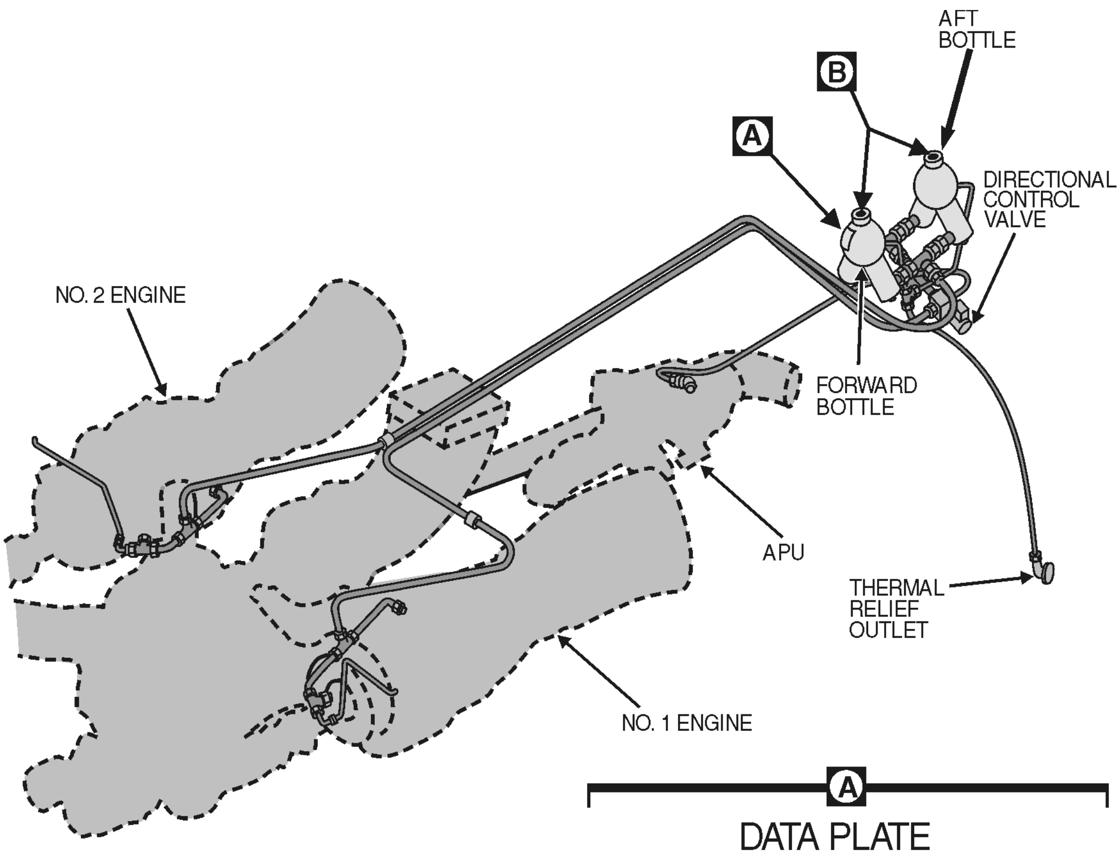
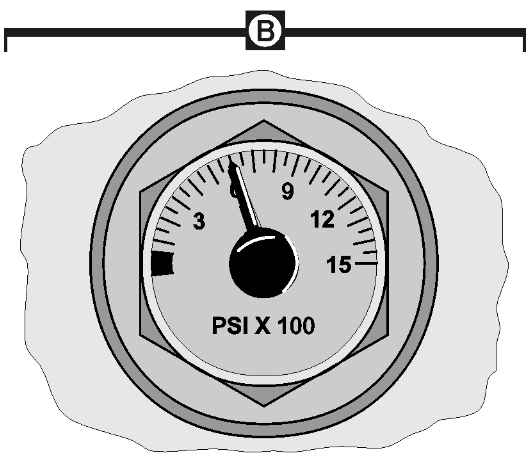


Figure 2-51. Fire–Detection System, Block Diagram

2.15 ENGINE/APU FIRE-EXTINGUISHING SYSTEM

The bromotrifluoromethane (CF_3 Br) high–rate discharge extinguishing system (Figure 2-52) provides a two–shot (main and reserve) capability to either the main engine compartments or the APU compartment. The system includes two containers that are filled with extinguishing agent and charged with nitrogen. The containers are mounted above the upper deck, aft of the APU compartment. Both containers have dual outlets, each outlet containing its own firing mechanism and CAD. Each container has a pressure gauge and a thermal discharge relief port. Thermal discharge is indicated by the loss of a red plastic disc on the left side of the aircraft. Electrical power to operate the system is supplied by the battery utility bus, the NO. 2 DC primary, and the DC essential bus through the FIRE EXTGH circuit breakers on the lower console, the overhead console, and the ATO circuit breaker panels. Three T–shaped handles select the compartment to which the fire extinguishing agent is to be directed and shut off fuel to that engine or APU. The FIRE EXT switch on the overhead console has three positions marked RESERVE, OFF, and MAIN. The MAIN position of the switch sends fire extinguishing agent to NO. 1 engine or APU compartment from the forward fire extinguishing bottle M1 port. The R2 port is reserve for NO. 2 engine. The aft fire extinguisher bottle M2 port is main for the NO. 2 engine and the R1 port is reserve for NO. 1 engine or APU. The ports provide a second shot of extinguishing agent to be used if the actuation is not enough and the bottle was not previously discharged. The fire extinguisher selector switch is armed after one of the T–handles has been pulled. If two T–handles are pulled, whichever T–handle is pulled last will be armed. When placed to MAIN or RESERVE, it selects the container to be discharged.





A

DATA PLATE
TEMP. - PRESSURE RANGE

°F	-40°	-20°	0°	+20°	+40°
PSIG	292	320	355	396	449
	370	400	437	486	540
°F	+60°	+70°	+80°	+100°	+120°
PSIG	518	555	593	670	775
	618	660	702	748	885

NOTE

- GAUGE 0-1500 PSI IN 50 PSI INCREMENTS
- SAME BOTH BOTTLES

SH60B-F048

Figure 2-52. Fire–Extinguishing System (Sheet 1 of 2)

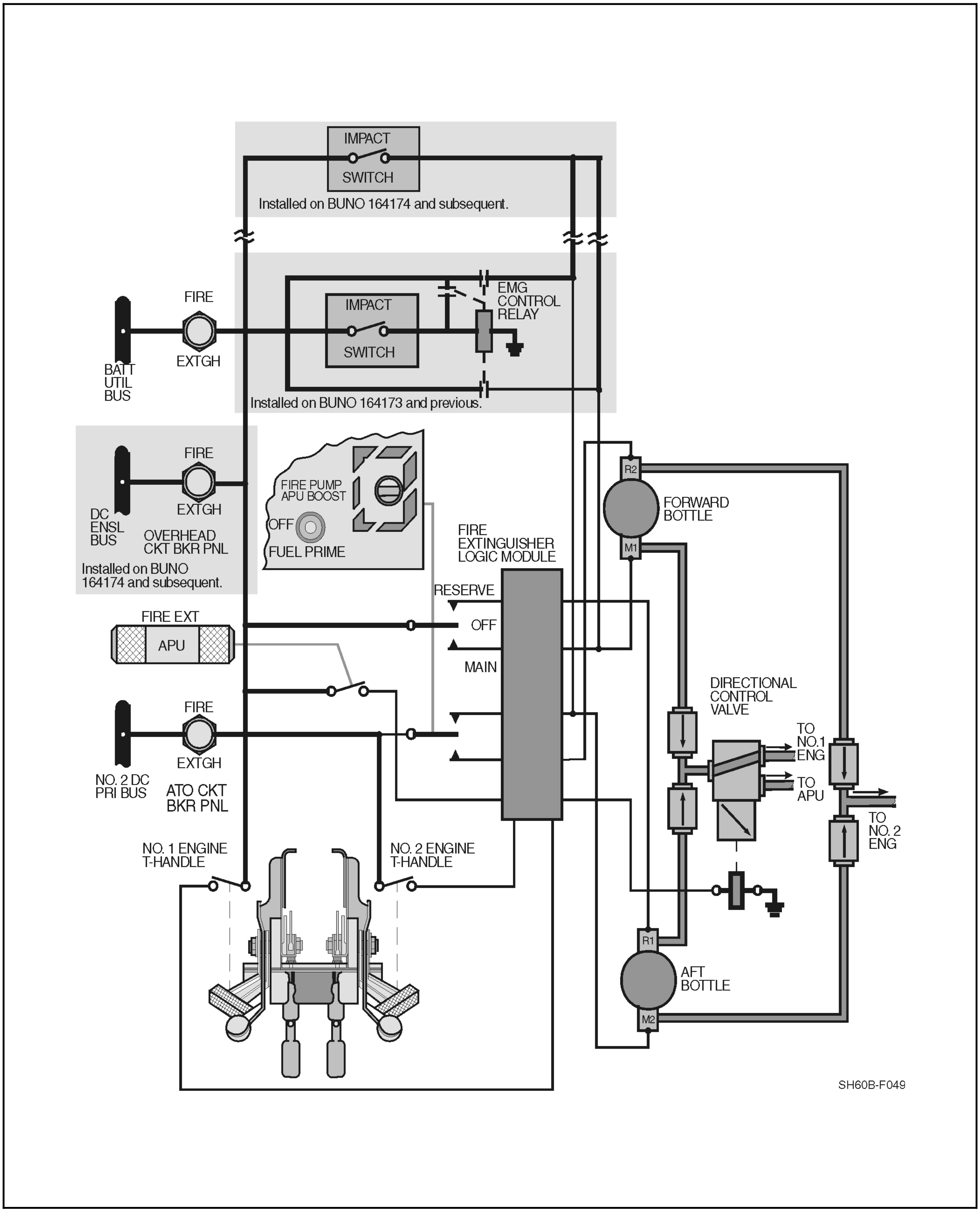


Figure 2-52. Fire-Extinguishing System (Sheet 2)

Note

- On aircraft BuNo 164173 and previous, if the NO. 2 DC primary bus is not energized (no AC power), the reserve position must be used to discharge the agent to the APU or NO. 1 engine compartment. The NO. 2 engine compartment has no fire extinguisher capability without AC power available.
- On aircraft BuNo 164174 and subsequent, if the NO. 2 DC primary bus is not energized (no AC power), the reserve position must be used to discharge the agent to the APU or NO. 1 or NO. 2 engine compartment.

2.15.1 Fire-Extinguishing System, Impact Switch Operation

A multiple–axis impact (10g) sensor (Figure 2-52), hard mounted to the airframe, will automatically discharge both fire bottles into both engine compartments when it senses crash forces. When the impact switch closes, power from the battery utility bus will be applied to the fire bottles, discharging the extinguishing agent into the NO. 1 and NO. 2 engine compartments. Electrical power for the impact switch is from the battery utility bus through a circuit breaker, marked FIRE EXTGH, on the lower console circuit breaker panel.

2.16 ENTRANCE AND EGRESS

A hinged door is located on each side of the cockpit. The sliding door on the right side of the cabin provides an opening 54 inches high by 44 inches wide. Emergency escape can also be accomplished through jettisonable features, provided on all cockpit and cabin windows.

Each cockpit door is equipped with a jettison system for emergency release of the window. Each window is jettisoned from inside or outside the cockpit by use of a handle marked EMERG EXIT — PULL. To provide emergency exit from the cabin, two jettisonable 24 inch by 24 inch windows are installed, one in the cabin door and the other at the SO station. To release the windows, a handle, marked EMERGENCY EXIT PULL AFT OR FWD, is moved in the direction of the arrow. The windows can then be pushed out. Exterior release of all windows is accomplished by a handle, below the window, marked PUSH TO RELEASE & TURN; Refer to Chapter 12 for additional information.

2.17 ENVIRONMENTAL CONTROL SYSTEM

Cabin, cockpit, nose bay, and transition section environments are controlled by the Environmental Control System (ECS), which provides both heating and air conditioning. The ECS consists of an air–cycle machine (ACM), bleed–air ducting, necessary controls and valves, water separator, distribution system, air inlet, and heat–exchanger exhaust duct. The engines or APU can serve as bleed–air sources for the ECS. Air source selection is accomplished by means of the AIR SOURCE ECS/START switch on the upper console. In the ENG position, engine bleed–air is selected as the air source. In the APU position, APU bleed–air is used as the air source; however, the APU will provide bleed air to the ECS regardless of the AIR SOURCE ECS/START switch position if the APU is on.

With the ECS on and the FLOW switch in NORM, maximum torque available is reduced by 4 percent per engine and fuel flow to each engine will increase by approximately 8 pounds per hour. With the TEMP rotary switch in HOT and OAT below 15 °C, maximum torque available is reduced by 5 percent per engine. With the FLOW switch in HIGH, maximum torque available is reduced by 7 percent per engine and fuel flow increases approximately 12 pounds per hour per engine

An overpressure switch, within the ECS, senses high air pressure. When an overpressure condition exists, the overpressure switch causes the ECS HI PRESS advisory to appear. System shutdown does not occur during an overpressure; the ECS components are capable of withstanding full bleed–air pressure.

2.17.1 ECS Control Panel

The ECS control panel, located on the lower console, contains three toggle switches and a rotary switch. The MODE toggle switch controls the ECS operating modes. In OFF, the system is secured. In AUTO, the temperature is set by the rotary temp switch, which can be dialed to any position between COLD to HOT. In MAN, temperature is set via a toggle switch labeled HOT and COLD, which is spring-loaded to an unlabeled, neutral position. Holding the switch in the HOT or COLD position causes the temperature control valve to move as long as the switch is held. The remaining toggle switch is labeled FLOW. The NORM position should be used for regular heating and cooling functions. The HIGH setting provides an increased volume of air and is used primarily for cooling components and environmental control.

Note

Use of the manual mode of the ECS requires pulsing of the HOT-COLD toggle switch followed by a waiting period to judge the magnitude of temperature change. Excessive manual input may cause ECS shutdown and/or APU failure.

The ECS will automatically shut down under the following conditions:

1. Engine contingency power is selected by either collective CONTGCY PWR switch.
2. NO. 1 or NO. 2 starter is engaged.
3. An ECS heating duct over-temperature exists.

When the AIR SOURCE ECS/START switch is placed to ENGINE, the ECS will also shut down when:

1. Actuation of IRP limiter (839 ±10 °C).
2. Either ENG ANTI-ICE switch is placed ON.
3. The DE-ICE MASTER switch is placed to AUTO and ice is detected.
4. An ECS underpressure situation exists.

ECS shutdown will be indicated by an ECS SHUTDOWN caution light in all of the above situations except engine start and ECS underpressure.

2.17.2 Avionics Cooling

The total aircraft avionics system requires the dissipation of approximately 12 kilowatts of heat. Units cooled by the external air system are maintained at 15 to 27 °C. Units cooled by ambient cabin air require an ambient temperature below 29 °C.

Two fans provide cooling air for the mission avionics. One fan is located on the right side of the cabin at the base of the mission avionics rack, and the other is located on the left side of the cabin at the base of the SO console. Fan control is provided by the mission power (MSN PWR) switch, located on the center console on the mission systems (MSN SYS) panel (Figure 1-7), and by a 27 °C temperature-sensing switch, located at each fan inlet. When the MSN PWR switch is placed in either PRI or SEC position and the fan inlet temperature is above 27 °C, the fans run to bring in outside air for circulation through the respective avionics areas. Backup cooling for the avionics is provided by the ECS. If the ECS is operating, the modulating valve will automatically go to the full-open position when the temperature switches at the fan inlets sense a temperature of 55 °C or greater. Conditioned cabin air may be circulated through the avionics system by removing the thermal/acoustic panels for backup cooling. Power is supplied from the NO. 1 AC primary bus and NO. 2 AC primary bus through the SO circuit breaker panel (Figure 2-22) by two circuit breakers marked LH RACK BLOWER and RH RACK BLOWER.

2.18 DE-ICE/ANTI-ICE SYSTEMS

2.18.1 DE-ICE MASTER Switch

The DE-ICE MASTER switch is on the overhead console (Figure 1-6). Placing this switch to AUTO with the ENG ANTI-ICE switches, WINDSHIELD ANTI-ICE switches, and the BLADE DE-ICE POWER switch at OFF will automatically turn these systems on when ice accumulation is sensed by the ice detector. Whenever the ice detector senses ice, the ICE DETECTED caution light will illuminate. Placing the DE–ICE MASTER switch to MANUAL disables the automatic function. In addition, placing the ENG ANTI–ICE switches, WINDSHIELD ANTI–ICE switches, or the BLADE DE–ICE POWER switch to the ON position with the DE–ICE MASTER switch at AUTO, disables the automatic function and the appropriate system will operate continuously.

2.18.2 Engine and Inlet Anti–Ice System

Refer to paragraph 2.1.8.

2.18.3 Rotor Blade De–Ice System

The rotor blade de–ice system (Figures 2-53, 2-54, 2-55 and 2-56) consists of the following: system control panel, test panel, system controller, power distributor, main and tail slip rings, main and tail blade heating elements, caution lights, outside air temperature (OAT) sensor, a modified ambient sense line and an ice detector/signal converter subsystem.

The blade de–ice system provides controlled electrical power to integral heating elements in the main and tail rotor blades, causing the ice bond layer to weaken, allowing symmetrical ice shedding. The blade de–ice system, excluding an element–on–time (EOT) failure, may be ground–checked with the use of external power. AC power, is supplied through the blade de–ice distributor.

2.18.3.1 BLADE DE–ICE Control Panel

The controls for operating the rotor blade de–ice system are on the BLADE DE–ICE control panel. Controls are described in Figure 2-54.

CAUTION

Leaving the blade DE–ICE power switch in the test position can lead to blade damage.

2.18.3.2 BLADE DE–ICE System Operation

The ice detector, mounted on the NO. 2 engine cowling, senses ice accumulation on a vibrating probe by measuring the change in probe frequency. When the ice detector senses an accumulation of ice, the ICE DETECTED caution will be illuminated. Simultaneously, an aspirator heater on the probe is turned on to heat the probe, shed the accumulated ice and reset it for another cycle. The severity of the icing environment is proportional to the rate at which the probe heater is cycled. If the BLADE DE–ICE POWER switch is turned on after the ICE DETECTED caution is illuminated, the caution will remain illuminated as long as there is ice. The OAT sensor, installed below the windshield, provides a signal to the controller to govern heating element on time (EOT). The lower the OAT the longer EOT will be. With the mode selector switch set to AUTO, the controller processes the ice rate signal to produce heater element–off–time, and the OAT signal to produce the heater EOT.

The controller then sends command signals through the main rotor slip rings to the system distributor. The system distributor then switches power in sequence to the main rotor blade heater zones. To reduce power requirements, the blades are deiced in cycles. Tail rotor blade power is switched directly by the controller and sent through the tail rotor slip rings to the tail rotor blades. A tail rotor blade distributor is not required since the power is applied to the four tail blades simultaneously (Figure 2-53).

During a single main generator failure, the AC Monitor bus, which contains the Main Rotor Blade De–Ice, will be dropped until the APU is started and the APU generator is placed on, picking up the AC monitor bus.

Droop stop heaters are provided for each of the four rotor head droop stops. The droop stop heaters supply heat to the droop stop pins and cams during icing conditions. The droop stops are continuously heated as long as the blade de–ice control panel power switch is in the POWER ON or TEST position.

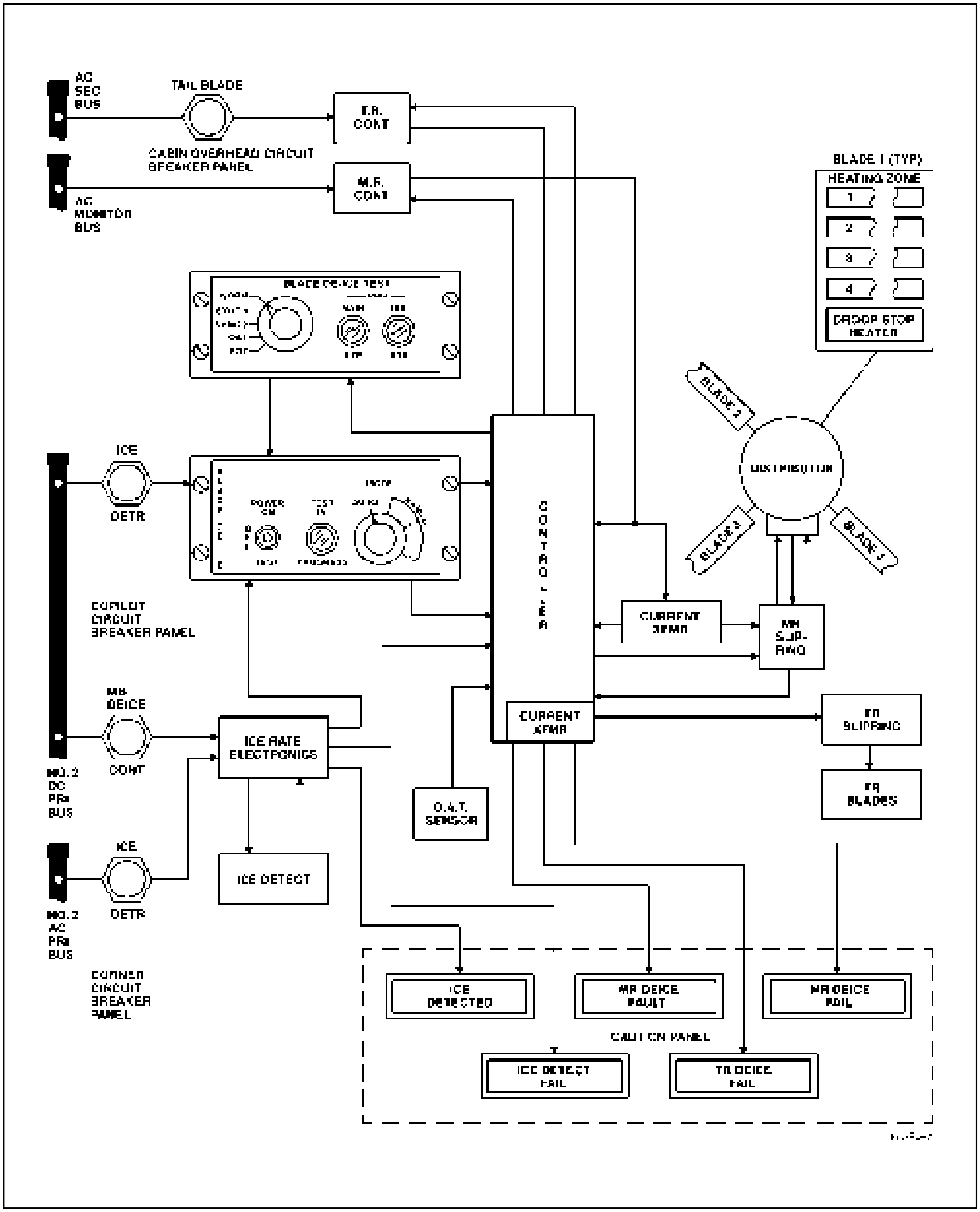


Figure 2-53. Blade De–Ice System Block Diagram

CONTROL	FUNCTION
POWER SWITCH:	
ON	Turns on power to blade de–ice system.
OFF	Turns off de–ice system.
TEST	Electrically tests main and tail rotor de–ice and signal convertor for one test cycle.
TEST IN PROGRESS LIGHT	Green light goes on during test cycle. At end of test cycle, light should go off.
MODE Selector:	
AUTO	System off-time is controlled by ice rate signal.
MANUAL	Gives pilot manual control of system off-time.
T	Trace.
L	Light.
M	Moderate.

Figure 2-54. Blade De–Ice Control Panel Functions

CONTROL	FUNCTION
NORM	Provides a signal path for normal operation.
SYNC 1	Provides a test signal to verify operation of Main Blade De–Ice synchronization short circuit warning circuitry when POWER switch is at TEST.
SYNC 2	Provides an open circuit to verify operation of Main Blade De–Ice synchronization open circuit warning circuitry when POWER switch is at TEST.
OAT	Short circuits the OAT sensor to check that BIT circuit senses a fault when POWER switch is at TEST.
EOT	Disables OAT sensor BIT circuits to simulate defects in primary EOT timing circuit, when POWER switch is ON and MODE select switch is at M (MODERATE).
PWR MAIN RTR light	Indicates a malfunction has occurred in the main rotor primary power when POWER switch is at OFF or ON. Also indicates test and normal operation when POWER switch is at TEST.
PWR TAIL RTR light	Indicates a malfunction has occurred in the tail rotor primary power when POWER switch is at OFF or ON. Also indicates test and normal operation when POWER switch is at TEST.

Figure 2-55. Blade De–Ice Test Panel Functions

The system control panel contains a rotary switch, which allows automatic or manual control of blade heater off time. In AUTO the ice rate signal is passed onto the controller, which results in off–time variations proportional to the icing rate. In MANUAL, (T, L, or M) fixed signals are transmitted to the controller resulting in fixed off time. One of the three manual modes should be selected when an icing rate system malfunction is indicated by the illumination of the ICE DETECT FAIL caution. The MANUAL mode should also be used when there is no indication of failure, but any of these three conditions has occurred:

1. The pilot has determined by judgment of icing intensity that the ice rate system is inaccurate.
2. Torque required has increased to an unacceptable level.
3. Helicopter vibration has increased to an unacceptable level.

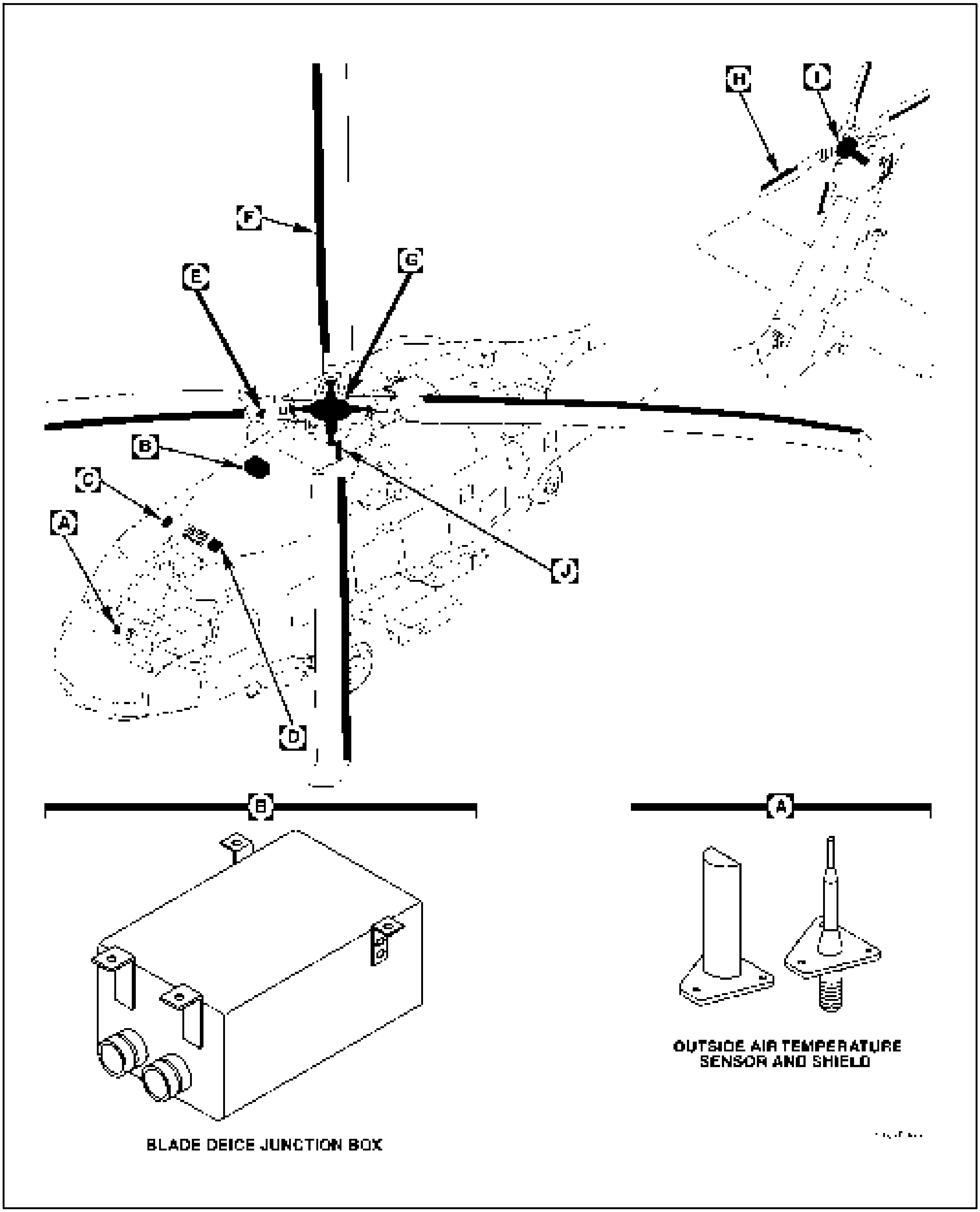


Figure 2-56. Blade De-Ice System (Sheet 1 of 2)