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BOOSTER SYSTEMS BRIEFS

BASIC, REV C

PREFACE

This document has been prepared by the Booster Section, Systems Division, NASA Lyndon B. Johnson Space Center, Houston, Texas, with technical support by Omniplan Corporation. Information contained in this document represents the Booster systems for STS missions as of October 1992. Revisions to this document will be made as required.

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1.1 SSME OVERVIEW

1.1.1 Introduction

The orbiter vehicle main engine propulsion system consists of three space shuttle main engines (SSME's). They are grouped in a cluster and are attached to the rear of the orbiter, aft of the payload bay compartment. The SSME's are reusable, high performance, liquid hydrogen/liquid oxygen-propellant rocket engines with variable thrust. Ignited on the ground at launch, they operate in parallel with the solid rocket boosters (SRB's) during the initial ascent phase, continuing for a total firing duration of approximately 520 seconds.

Each of the rocket engines operates at a mass mixture ratio (lb liquid oxygen/lb liquid hydrogen) of 6:1 and a main combustion chamber pressure of approximately 3000 psia. A sea level thrust of 375,000 lb and a vacuum thrust of 470,000 lb is the rated power level (RPL). The engines are can be throttled over a thrust range of 65 to 104 percent of the RPL. Full power level (FPL), 109 percent of RPL, can be commanded via item entry on SPEC 51, the override crew CRT display. The engines are gimballed (±10.5° for pitch and ±8.5° for yaw) to provide pitch, yaw, and roll control during the orbiter boost phase.

Significant in meeting performance requirements is a staged-combustion power cycle with high combustion chamber pressures. RPL operation is at a main combustion chamber pressure of approximately 3000 psia. In the SSME staged-combustion cycle, the propellants are partially burned at high pressure (5000 psia) and relatively low temperature in the preburners, then completely burned at high temperature and pressure in the main chamber before expanding through the high-area-ratio nozzle. The nozzle has an expansion ratio of 77.5 to 1. The specific impulse performance of about 452.5 seconds is greater than any previous large thrust rocket engine. Hydrogen fuel is used to cool all combustion devices directly exposed to high-temperature combustion products. Engine weight is minimized (less than 7000 lb) by using a welded type of construction wherever possible while still maintaining access to the internal areas of the engines for inspection and maintenance. An electronic engine controller automatically performs engine checkout, start, mainstage, shutdown, and post-shutdown functions.

1.1.2 SSME Numbering System

1.1.2.1 Development Engines

Engines are numbered 0001, 0002, 0003, 0004, etc.; 0001 being the Integrated Systems Test Bed (ISTB). The number is intended for single engine test firings. Each time a development engine is recycled for complete overhaul, its number is changed as follows: Engine 00XX becomes 01XX, then 02XX, 03XX, etc.

1.1.2.2 Production Engines

Engines are numbered 2001, 2002, 2003, 2004, 2005, etc. Each time a production engine is recycled for complete overhaul, its number is changed as follows: Engine 20XX becomes 21XX, then 22XX, 23XX, etc.

1.1.3 Schematics

Systems briefs 1.2 through 1.15 contain descriptions of flows, hardware, software, redlines, and controls of the SSME systems. The following drawings are overviews of the main engine propulsion system.

Figure 1.1-1.- Orbiter main propulsion system.

Figure 1.1-2.- SSME propellant flow schematic.

Figure 1.1-3.- SSME major components.

Figure 1.1-4.- SSME component location (looking aft).

Figure 1.1-5.- SSME installation orientation and interfaces (looking aft).

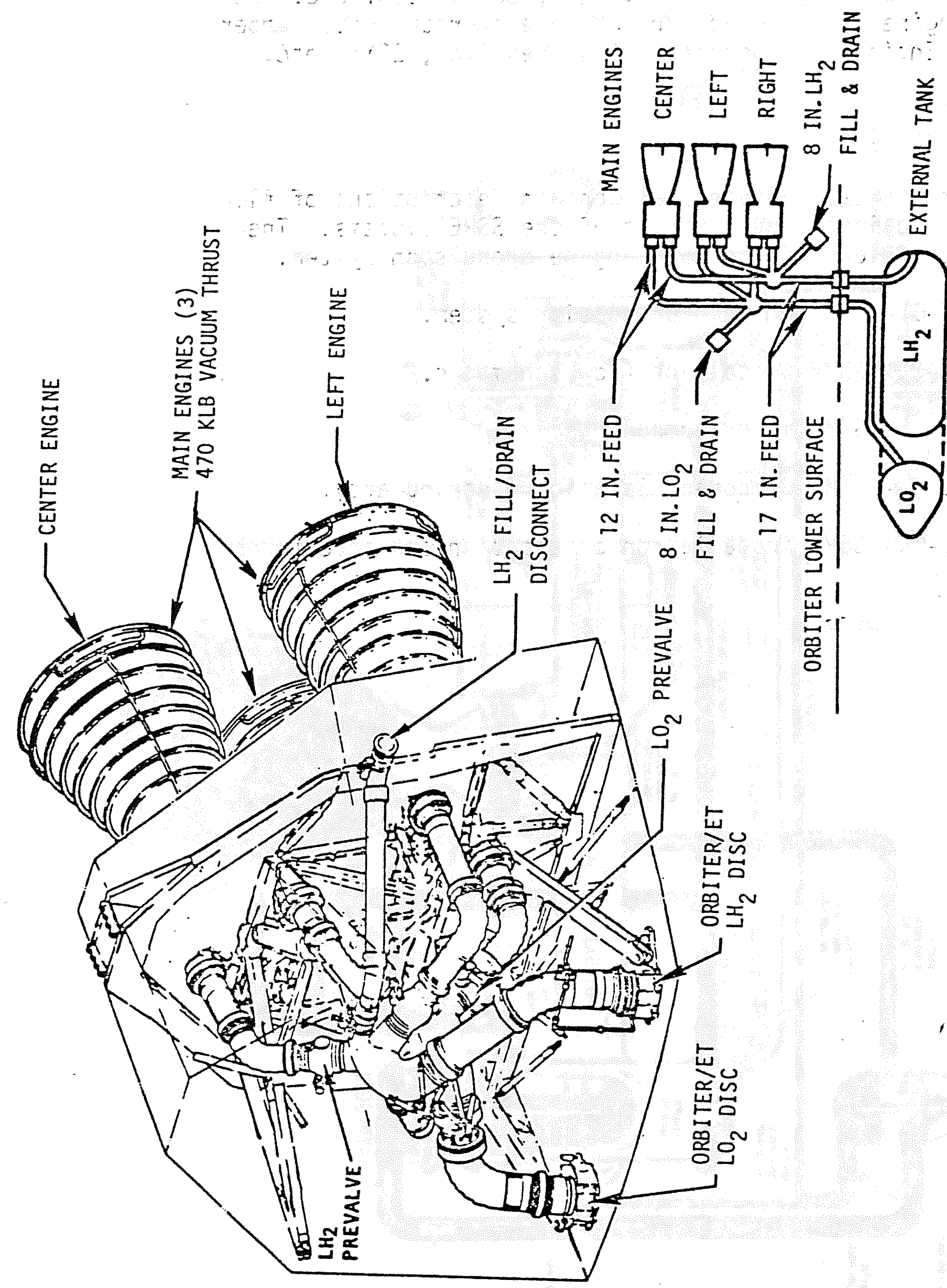


Figure 1.1-1.- Orbiter main propulsion system.

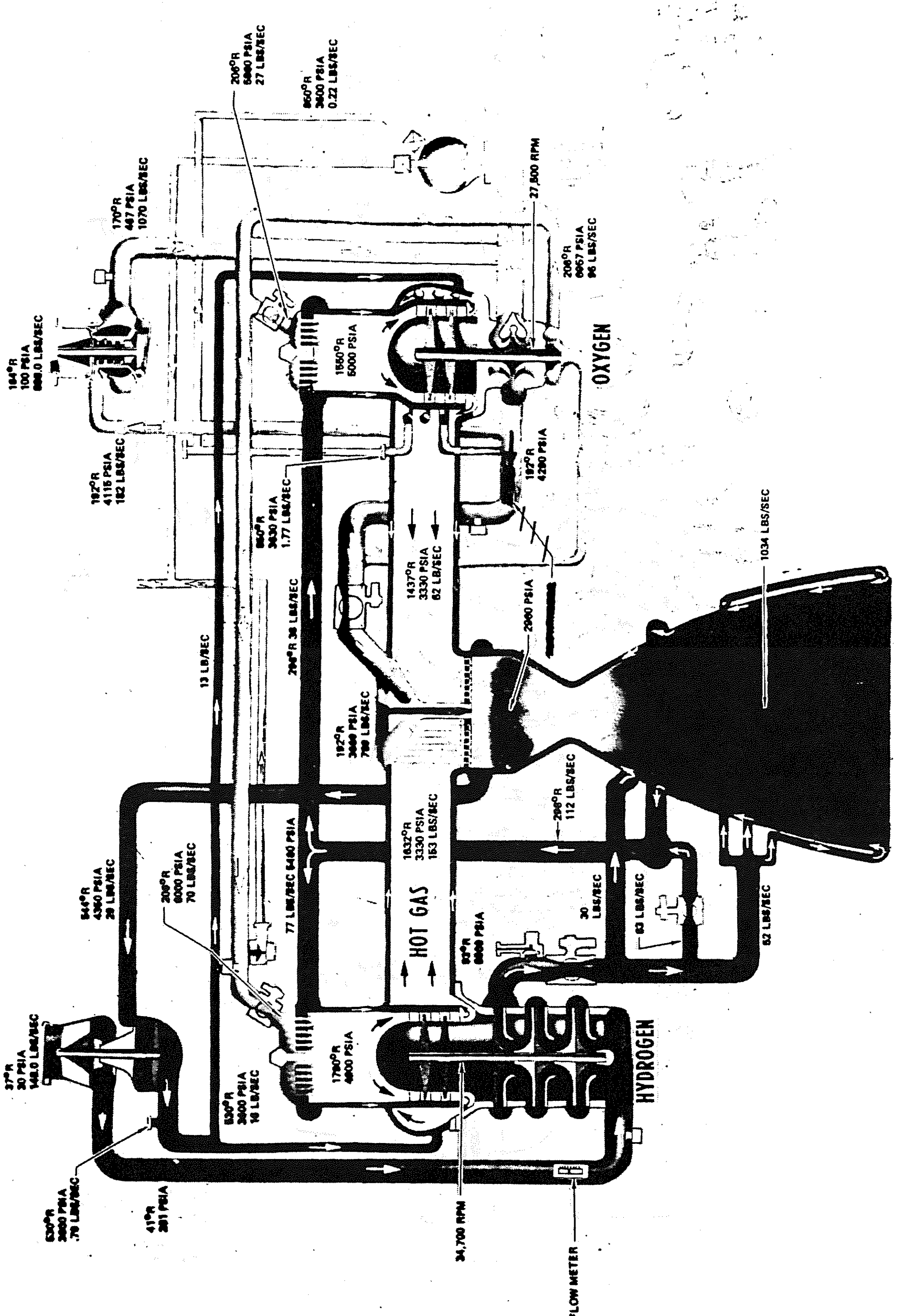


Figure 1.1-2.- SSME propellant flow schematic.

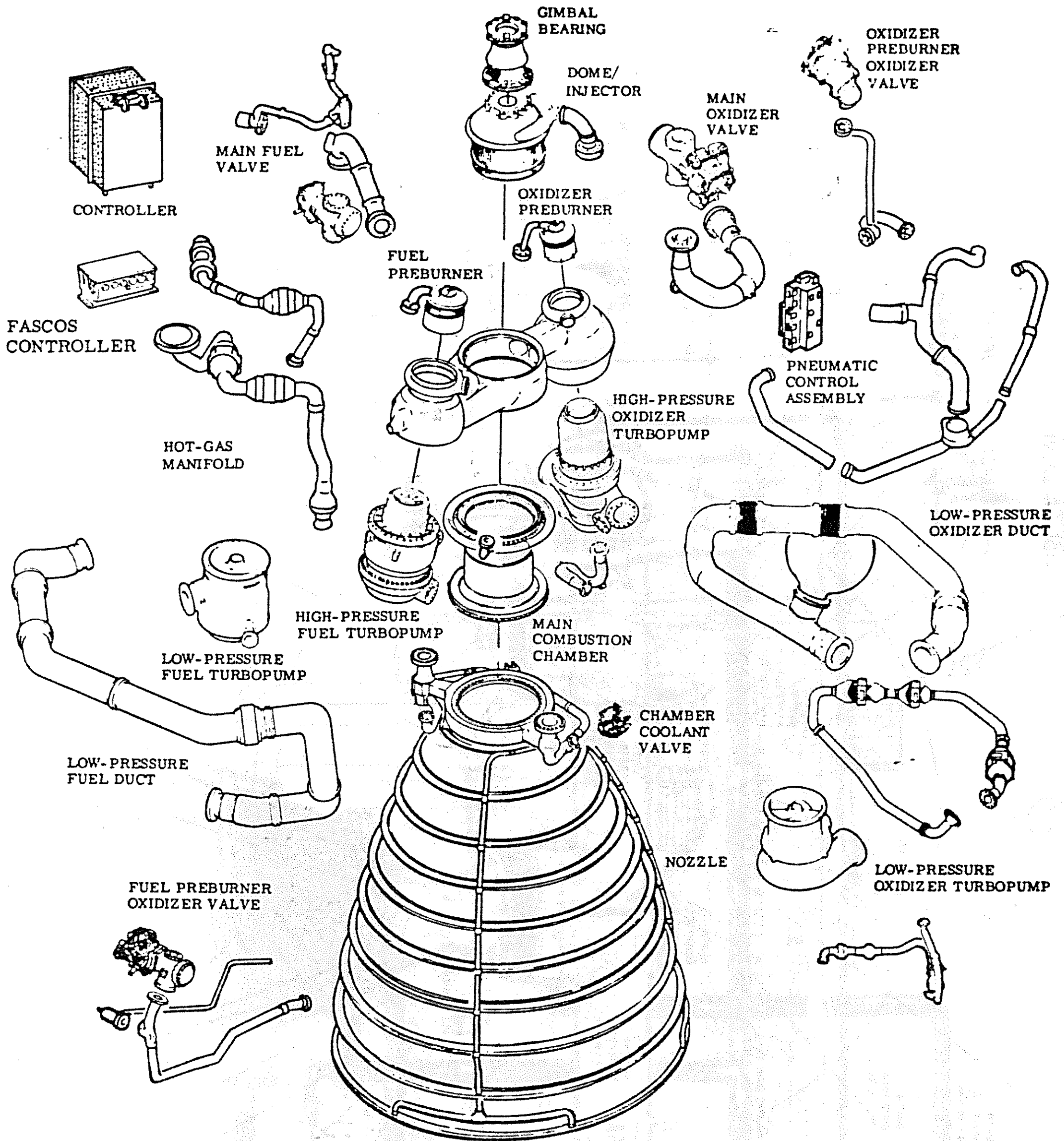


Figure 1.1-3.- SSME major componets.

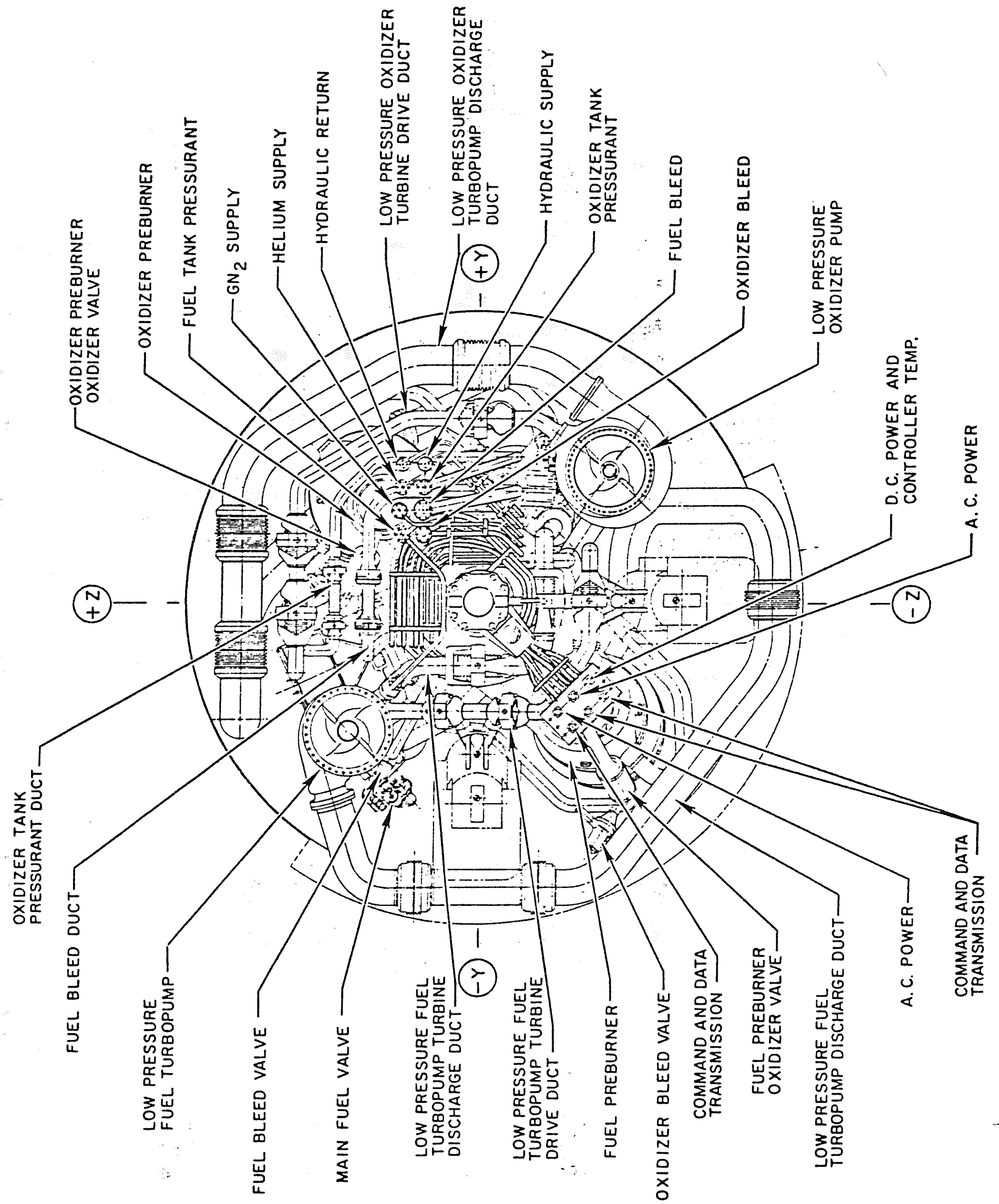


Figure 1.1-4.- SSME componet location (looking aft).

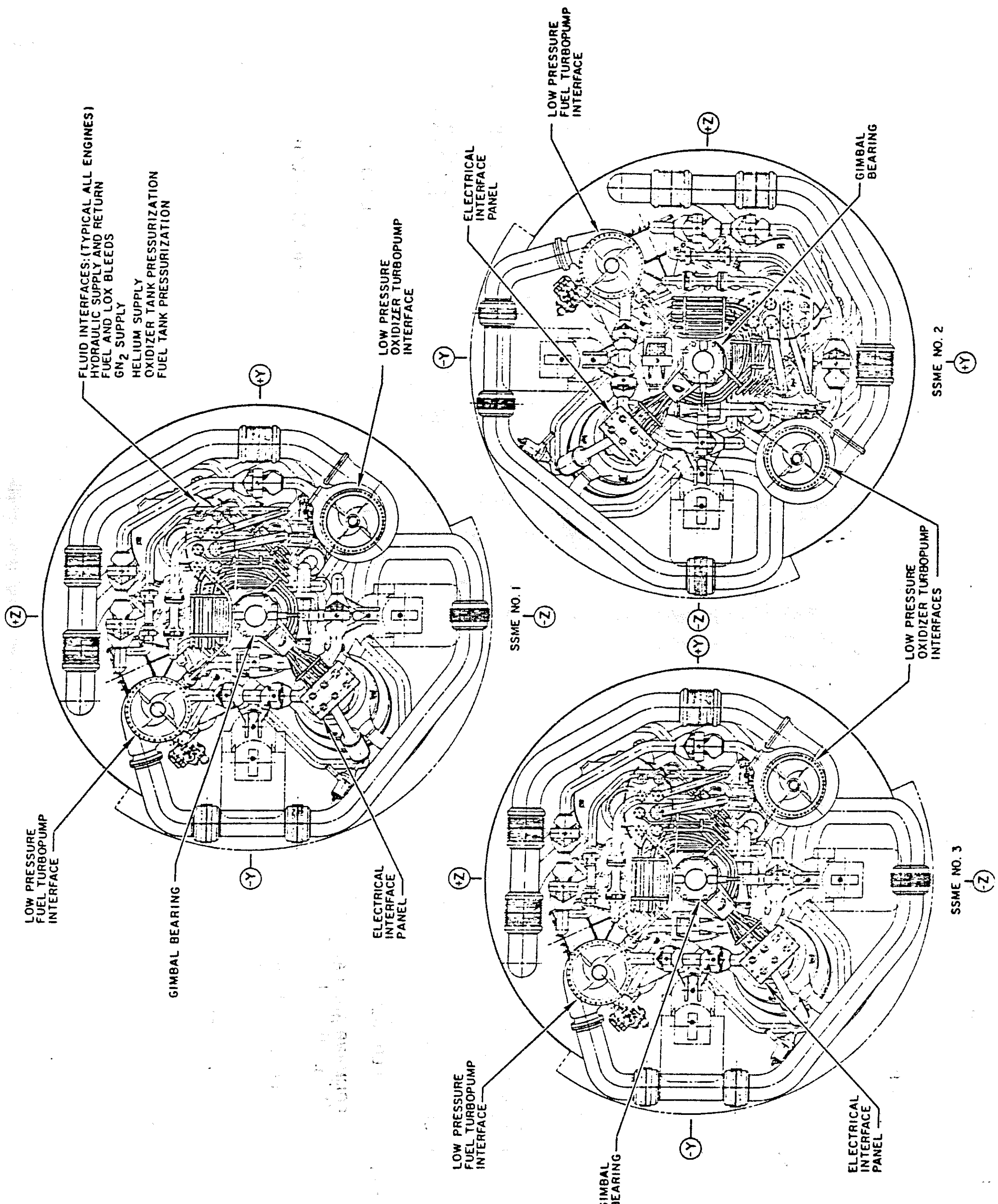


Figure 1.1-5.- SSME installation orientation and interfaces.

1.2 SSME PROPELLANT FLOW

1.2.1 Overview/General Description

The SSME uses a staged combustion cycle in which propellants are partially burned at high pressures and relatively low temperatures in the preburners and then completely burned at high pressure and temperature in the main combustion chamber (MCC) (fig. 1.2-1).

The propellant system uses four turbopumps. The two low-pressure turbopumps operate at relatively low speed to allow low ullage pressures in the propellant tanks. The function of these pumps is to provide sufficient pressure increase at the inlets of the high pressure turbopumps to permit them to operate at high speeds. The discharge from the low-pressure turbopumps is fed to the inlets of the high pressure turbopumps.

Approximately 75 percent of the flow from the high-pressure oxidizer turbopump (HPOT) goes to the MCC. Approximately 10 percent is directed to the preburner pump, which raises the pressure to that required by the preburners. Small quantities are bled through the heat exchanger for oxidizer tank pressurization and POGO suppression. The balance of the oxidizer drives the turbine powering the low-pressure oxidizer turbopump (LPOT) and is then recirculated to the inlet of the HPOT.

Approximately 20 percent of the high-pressure fuel turbopump (HPFT) discharge flow is used to cool the MCC, drive the low-pressure fuel turbopump (LPFT) turbine, cool the hot-gas manifold (HGM) and injector, and provide fuel tank pressurant. The remaining fuel is used first to cool the nozzle and then to supply the preburners.

The hot gas (hydrogen-rich steam) from the fuel and oxidizer preburners drives the high pressure pump turbines; then the steam flows to the main injector where it is mixed with additional oxidizer and fuel and injected into the MCC.

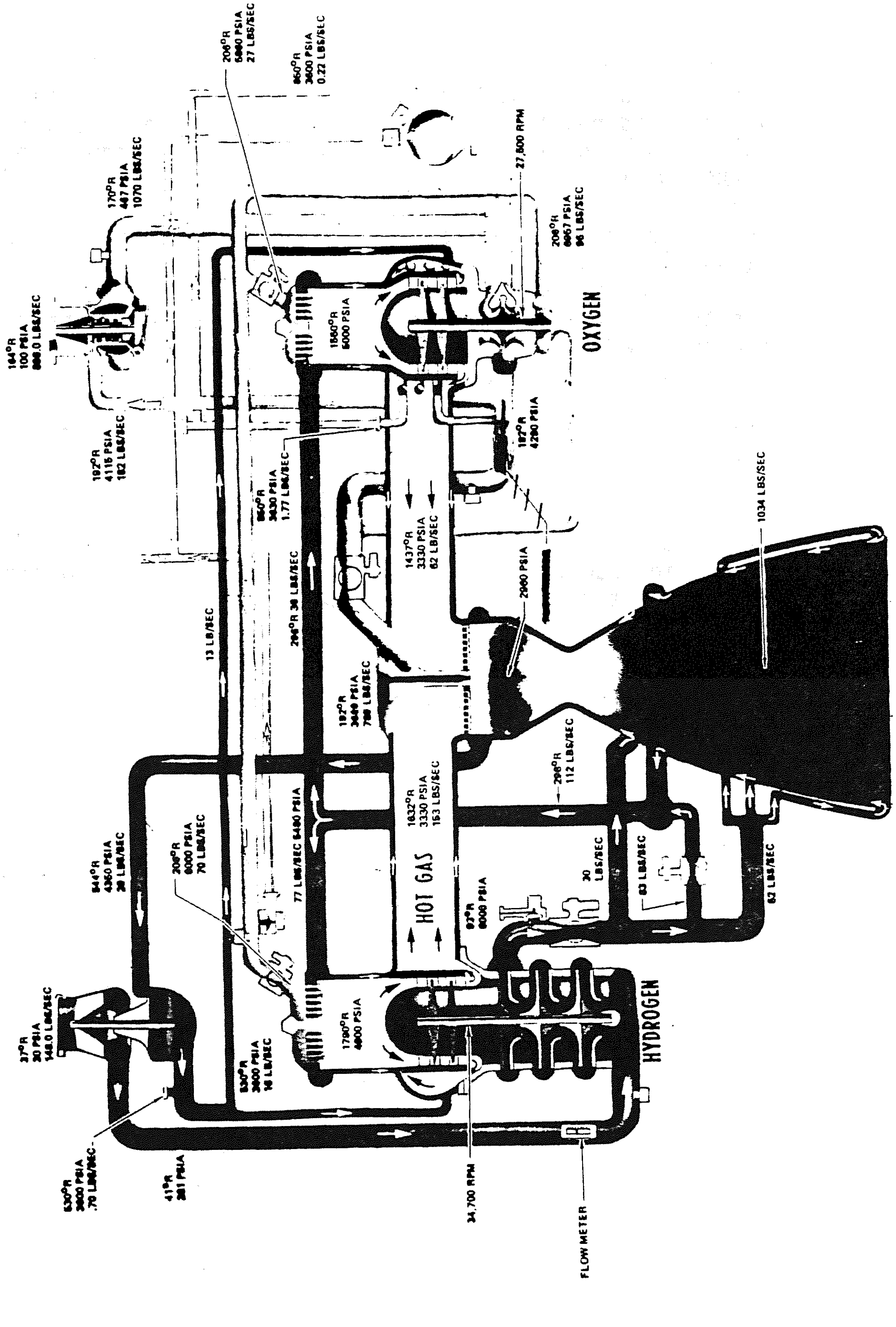


Figure 1.2-1.- The SSME propellant flow schematic.

1.2.2 Propellant Flow System Components

1.2.2.1 Hot-Gas Manifold

The HGM conducts hot gas (hydrogen-rich steam) from the turbines to the main injection elements of the main chamber injector. The area between the wall and the liner provides a coolant flow path for the hydrogen gas that exhausts from the LPFT turbine, protecting the outer wall and liner against the temperature effects of the hot gas from the preburners. This hydrogen, after cooling the manifold, is the fuel for the injector baffle elements and also the coolant for the primary faceplate, the secondary face, and the MCC acoustic cavities (fig. 1.2-2).

The oxidizer side of the HGM has a canted flange to which the HPOT is stud-mounted. Heat exchanger tube supports are welded to the inner wall on the oxidizer side of the HGM. Two hot-gas transfer tubes route the HPOT turbine exhaust gas to the main injector torus manifold, where it is radially directed into the hot-gas cavity of the main injector. The oxidizer preburner (OPB) is welded to the upper end of the oxidizer side of the HGM.

The fuel side of the HGM also has a canted flange to which the HPFT is stud-mounted. Three hot-gas transfer tubes route the HPFT turbine exhaust gas to the main injector torus manifold, where it is radially directed into the hot gas cavity of the main injector. The fuel preburner (FPB) is welded to the upper end of the fuel side of the HGM.

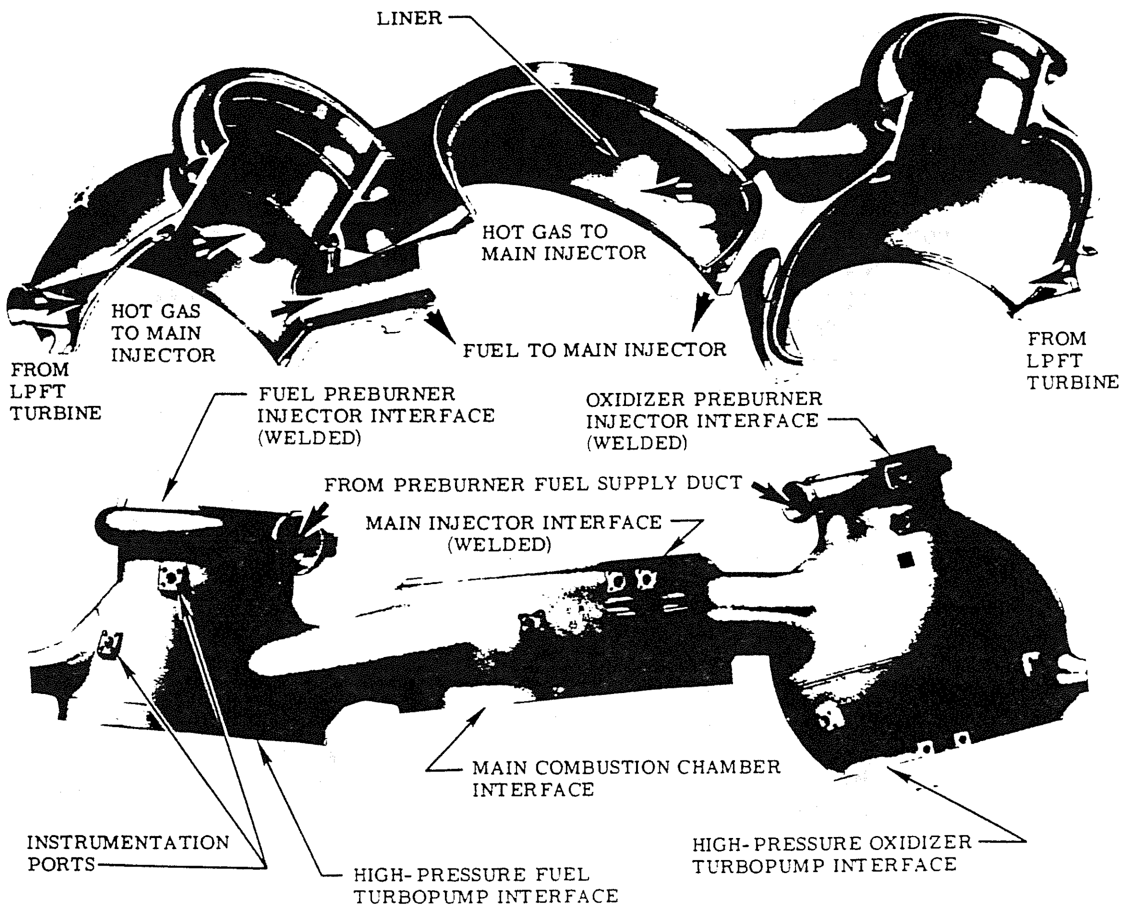


Figure 1.2-2.- Hot gas manifold.

1.2.2.2 Preburners

Two preburners burn hydrogen and oxygen to generate a variable hot gas supply to power the high-pressure turbopumps. They operate at a low mixture ratio with gaseous hydrogen from the nozzle coolant circuit combined with hydrogen bypassed through the chamber coolant valve and liquid oxygen from the preburner oxidizer pump. Specific operating levels of the preburners are controlled by regulating the oxidizer flow rate with the preburner oxidizer valves. There are both a fuel preburner (fig. 1.2-3) and an oxidizer preburner (fig. 1.2-4).

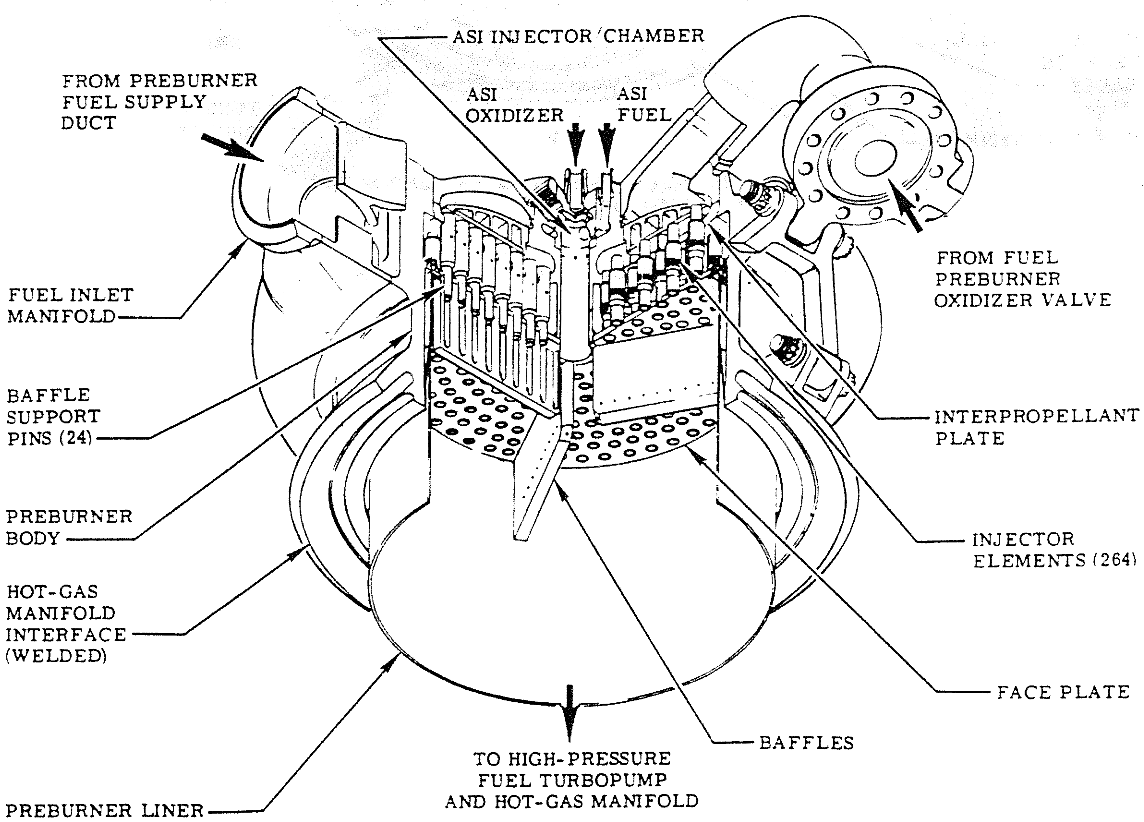


Figure 1.2-3.- Fuel preburner.

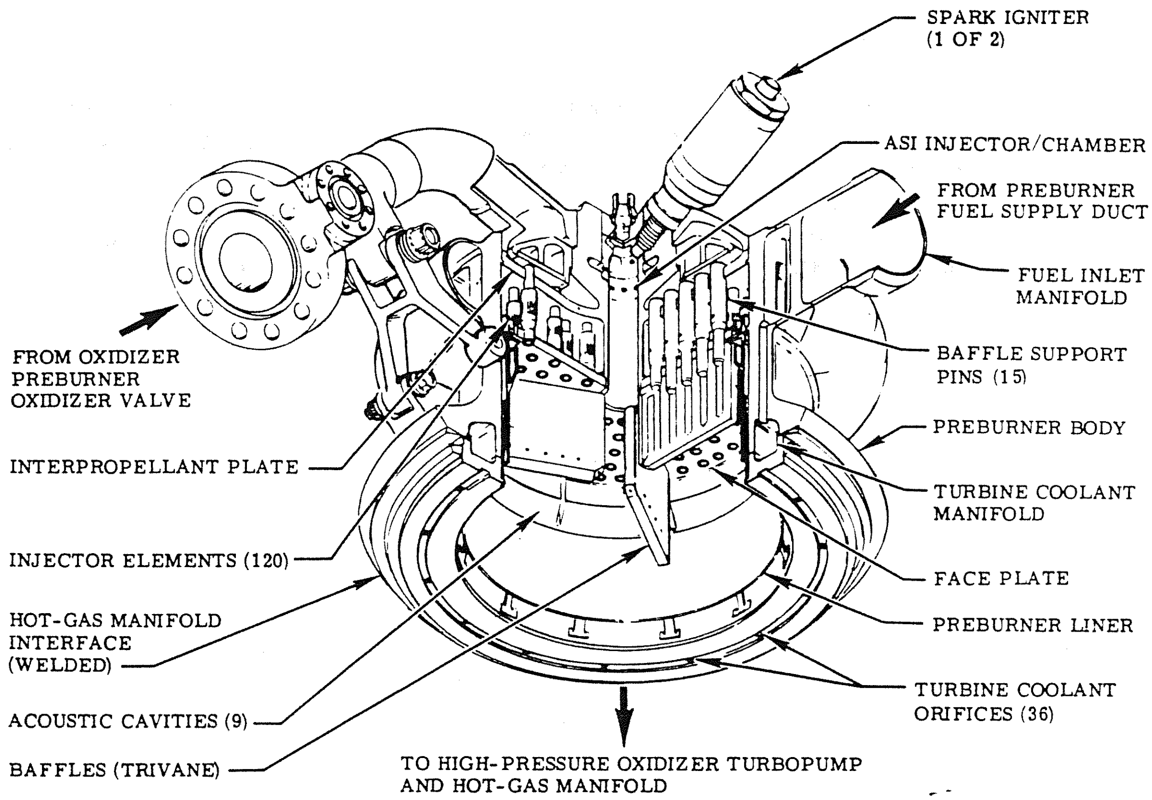


Figure 1.2-4.- Oxidizer preburner.

1.2.2.3 Main Injector

The main injector receives liquid oxygen from HPOT (via the high pressure oxidizer duct and the main oxidizer valve), receives gaseous hydrogen from the hot-gas manifold coolant circuit, and receives fuel-rich hot gas from the fuel and oxidizer preburners (via the hot-gas manifold). The main injector then efficiently mixes and uniformly distributes propellants to the MCC (fig. 1.2-5) for additional combustion.

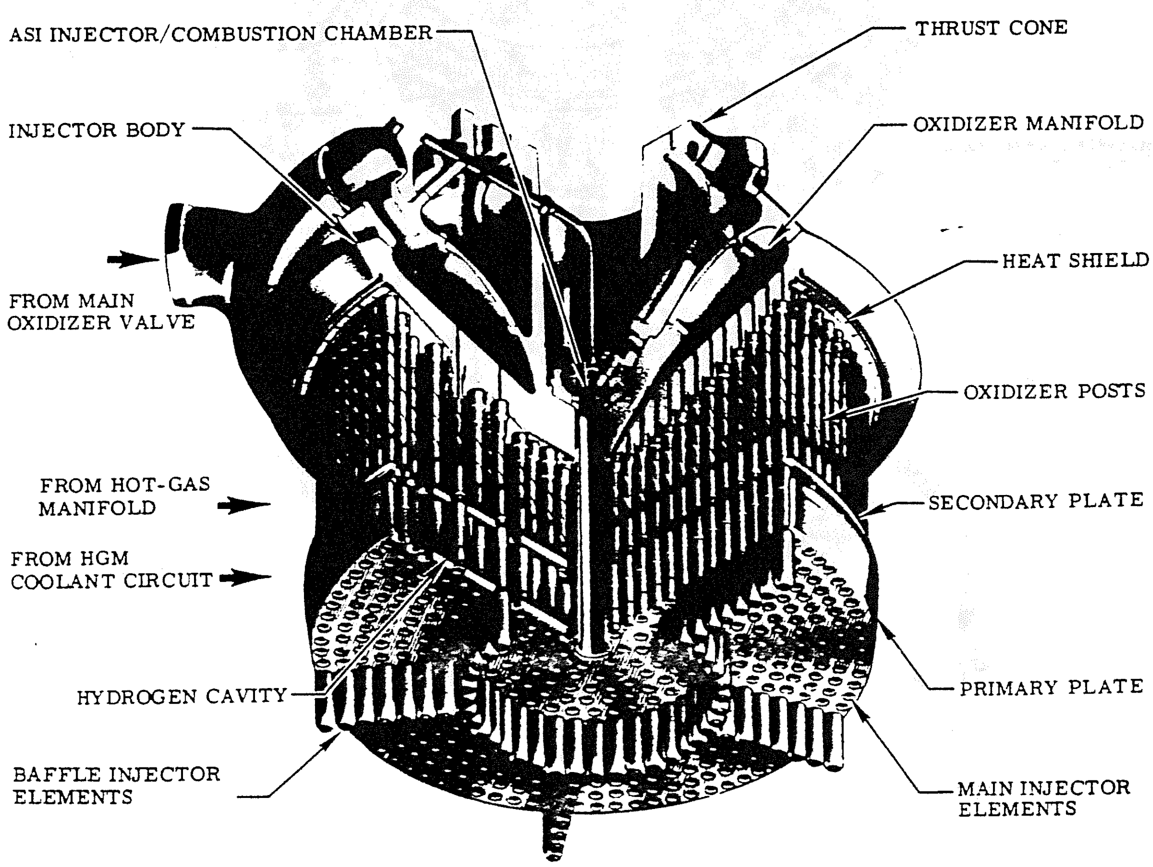


Figure 1.2-5.- Main injector.

1.2.2.4 Main Combustion Chamber

The MCC receives propellants from the main injector, contains the burning propellant gases, and initiates their expansion from the chamber throat to a ratio of 5:1. The chamber coolant liner provides the coolant flow path for the MCC. The channels are ported to coolant inlet and outlet manifolds of the chamber jacket to provide an up-pass circuit for MCC fuel coolant (fig. 1.2-6).

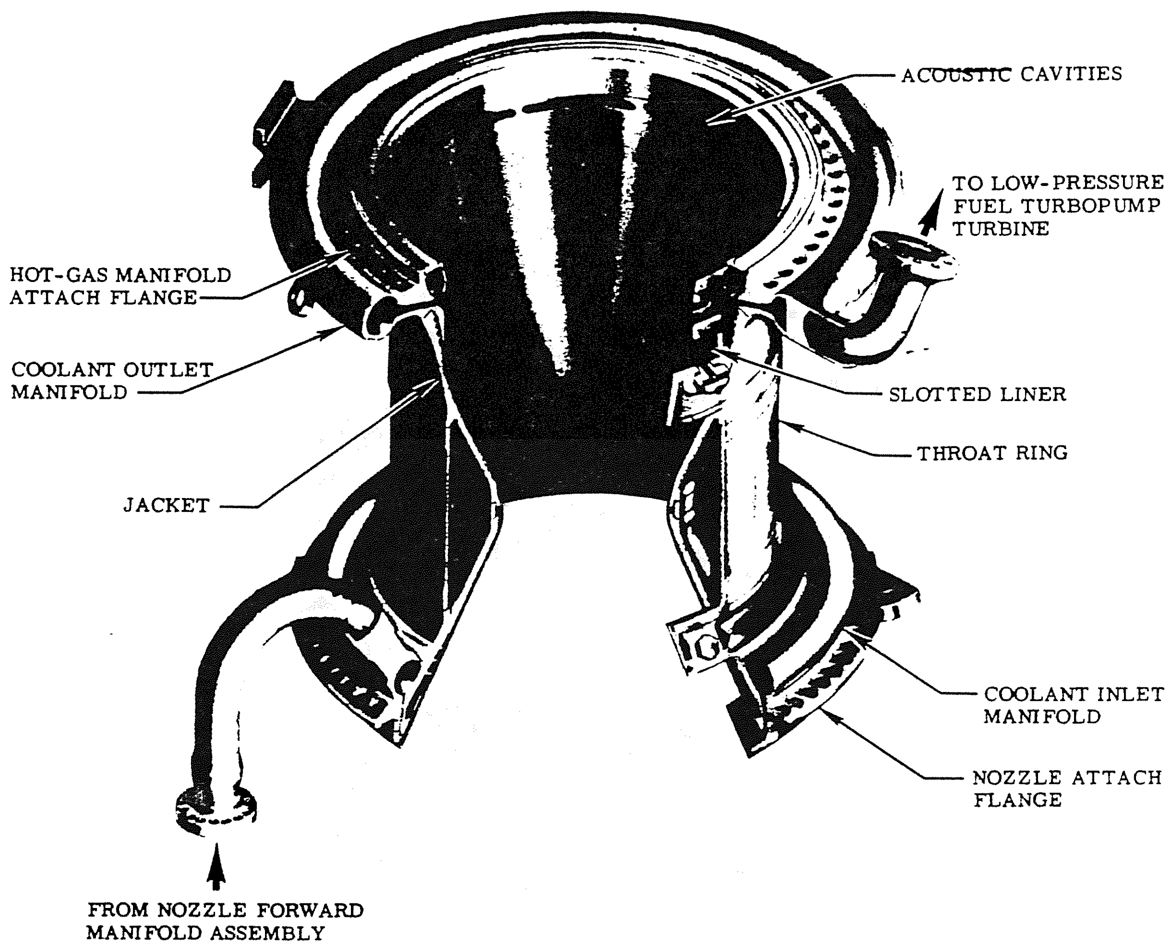


Figure 1.2-6.- Main combustion chamber.

1.2.2.5 Nozzle Assembly

The diffuser inlet interfaces with the downstream end of the main fuel valve (MFV) and distributes hydrogen to the fuel circuits. The diffuser directs fuel to the nozzle coolant inlet manifold, the MCC coolant inlet manifold, the chamber coolant valve (CCV), and the augmented spark igniters (ASI's). The mixer mixes fuel from the CCV with fuel from the nozzle coolant outlet and directs it to the OPB and FPB.

The nozzle subassembly provides an up-pass cooling circuit between the inlet and outlet manifolds. The hydrogen through the circuit is directed to a mixer where it joins the hydrogen flow through the CCV and is used as the fuel for the preburners (figs. 1.2-7 and 1.2-8).

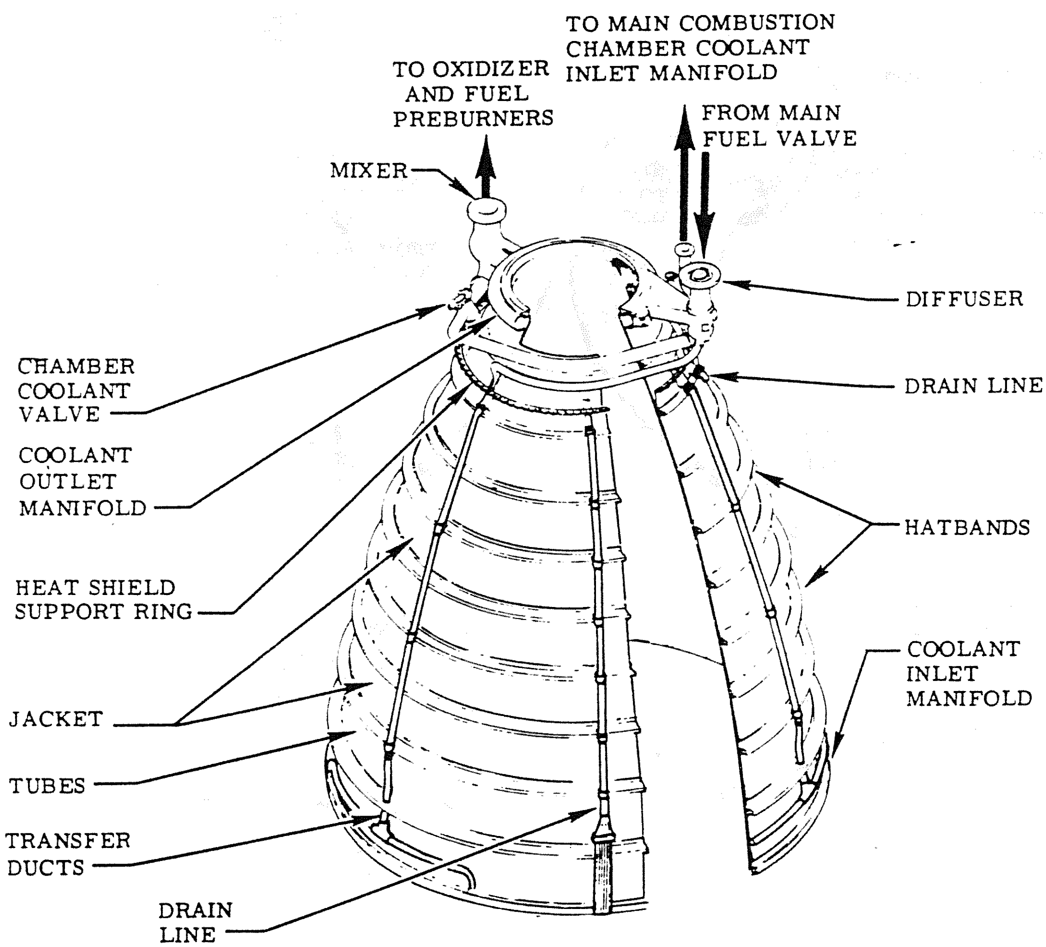


Figure 1.2-7.- Nozzle assembly.

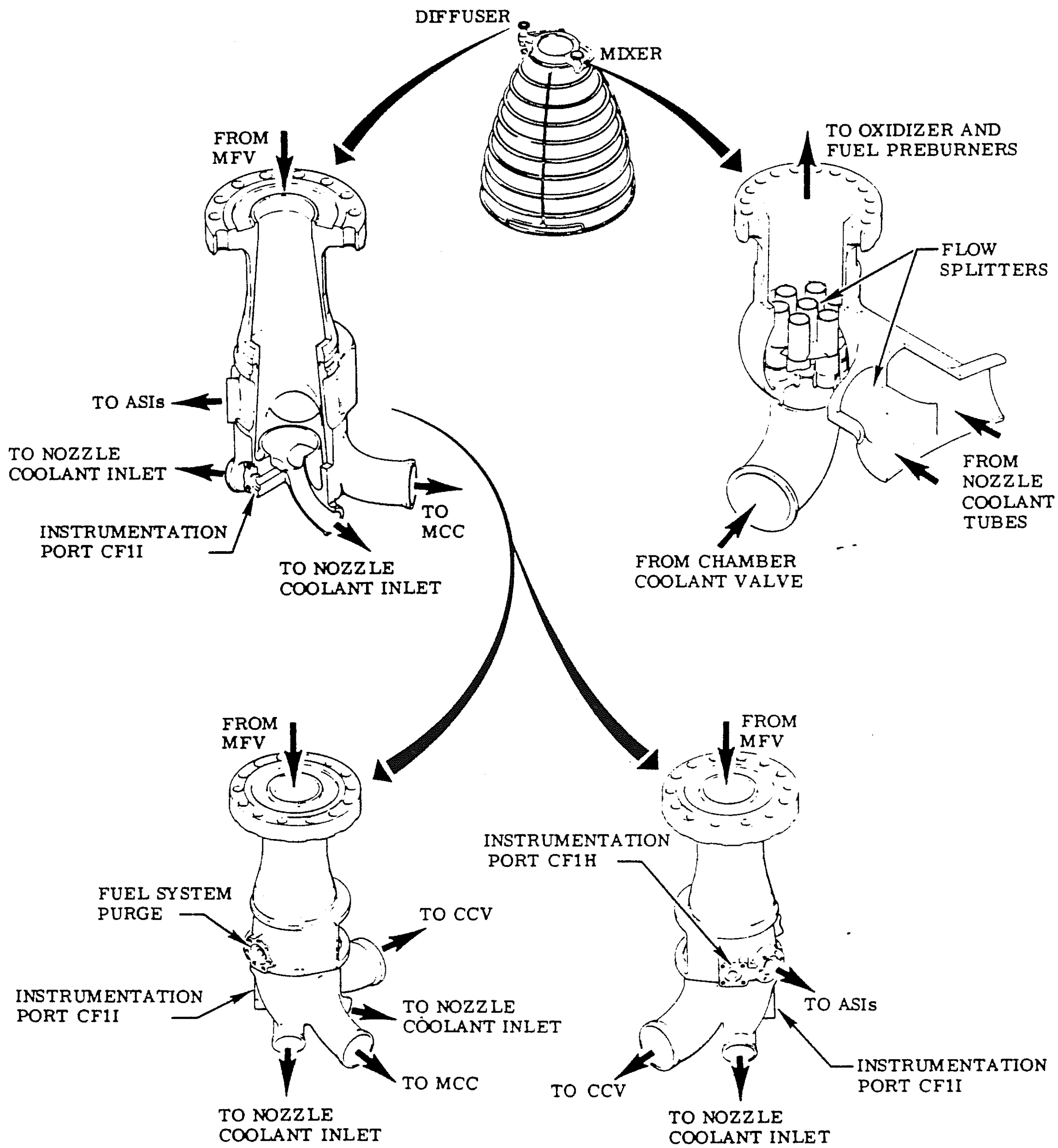


Figure 1.2-8.- Nozzle diffuser and mixer.

1.2.2.6 Low-Pressure Oxidizer Turbopump

The LPOT is powered by liquid oxygen (fig. 1.2-9). During engine start and mainstage, the LPOT maintains sufficient pressure to the HPOT to permit the HPOT to operate at high speeds. Turbine-drive fluid is tapped from the HPOT discharge and, after powering the turbine, is injected into the pumped fluid through a port between the turbine discharge and pump discharge volutes. The combined flows are then routed to the HPOT inlet.

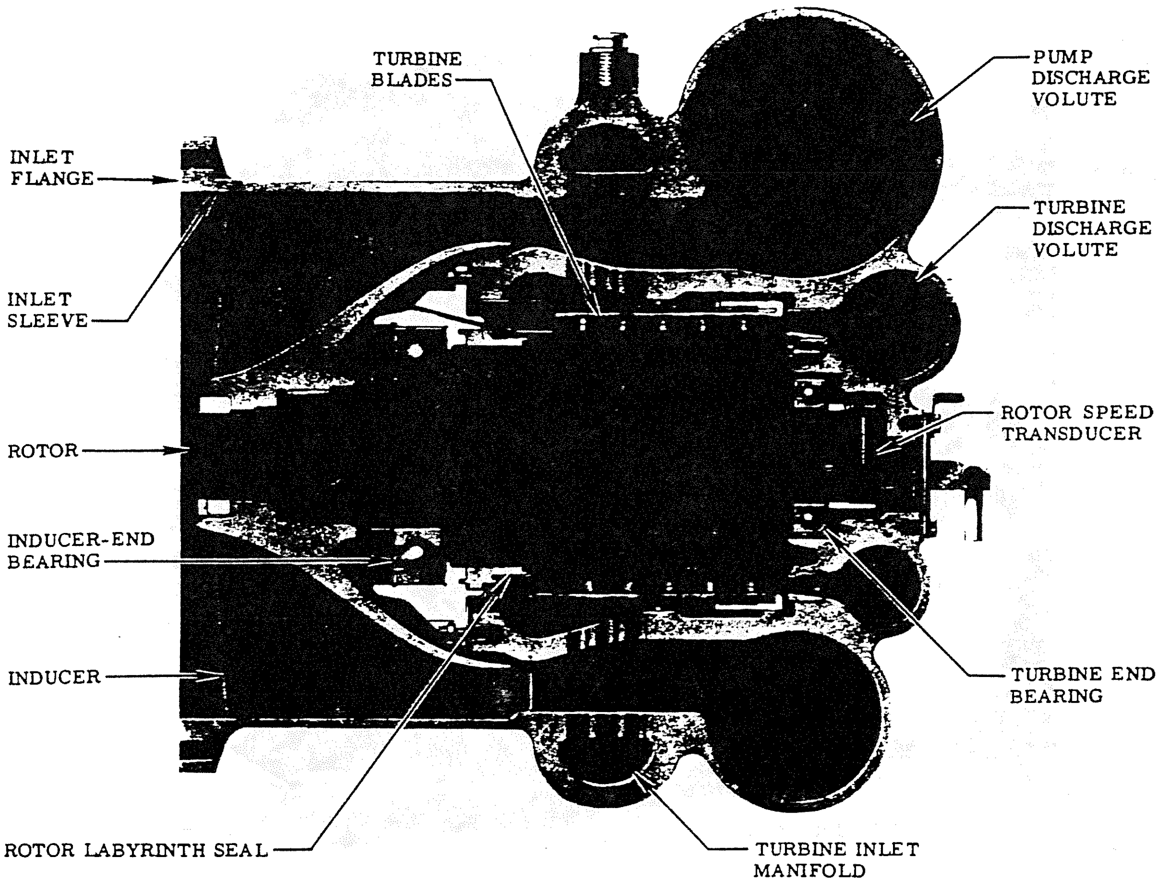


Figure 1.2-9.- Low-pressure oxidizer turbopump.

1.2.2.7 Low-Pressure Fuel Turbopump

The LPFT uses gaseous hydrogen (GH2) as the power medium (fig. 1.2-10). During engine start and mainstage operation, the LPFT maintains sufficient pressure to the HPFT to permit the HPFT to operate at high speeds. The turbine is driven by GH2 from the MCC coolant valve manifold. The LH2 coolant flows through the pump end bearings and turbine bearing. Then the coolant is returned to the pump inlet through passages in the shaft, bearing spacer, and inducer.

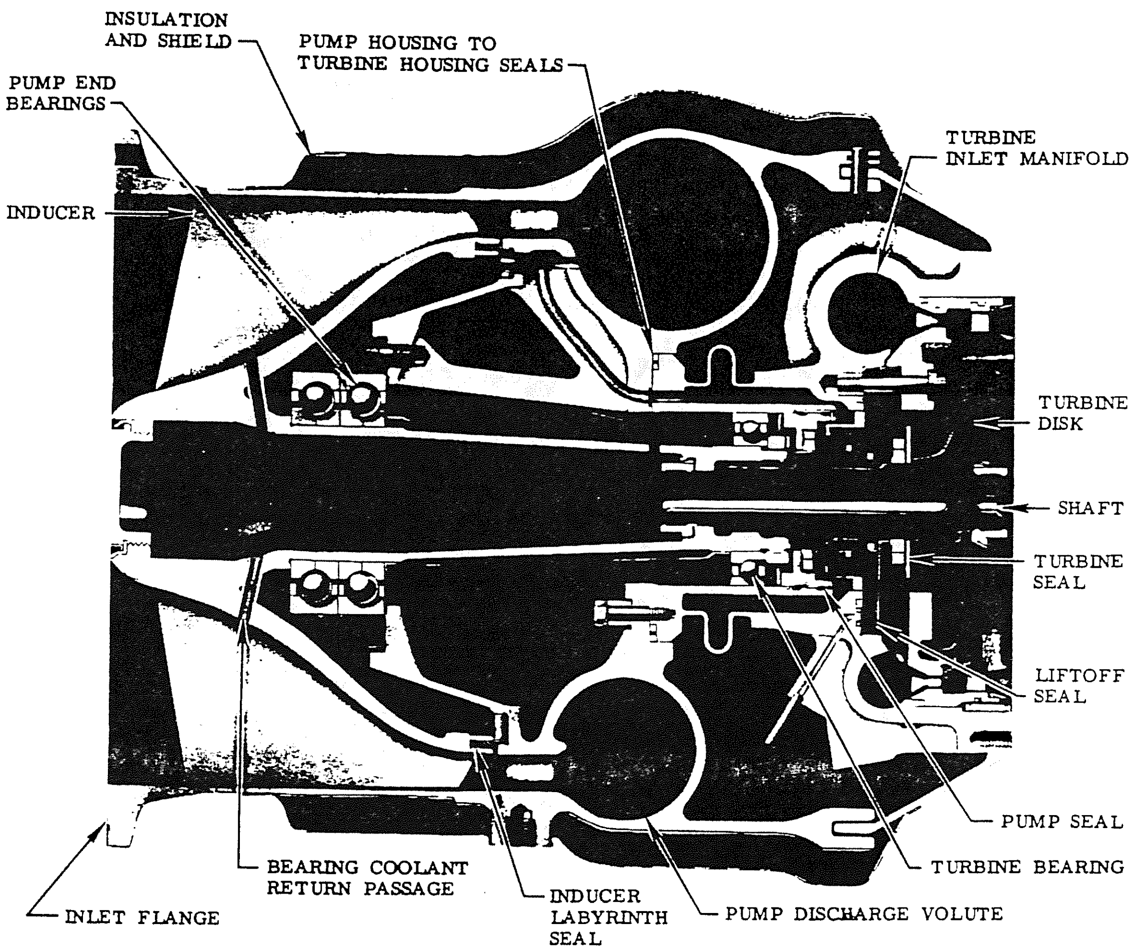


Figure 1.2-10.- Low-pressure fuel turbopump.

1.2.2.8 High-Pressure Oxidizer Turbopump

The HPOT is directly driven by a hot-gas turbine (fig. 1.2-11). The HPOT main pump has a single inlet with 50-50 flow split into a double-entry, common outlet impeller. Liquid oxygen enters the main pump through the main pump housing, where the flow split is made. Inlet vanes direct the flow to the impeller inlets. After passing through the impeller, the flow is redirected into the discharge volute by diffuser vanes.

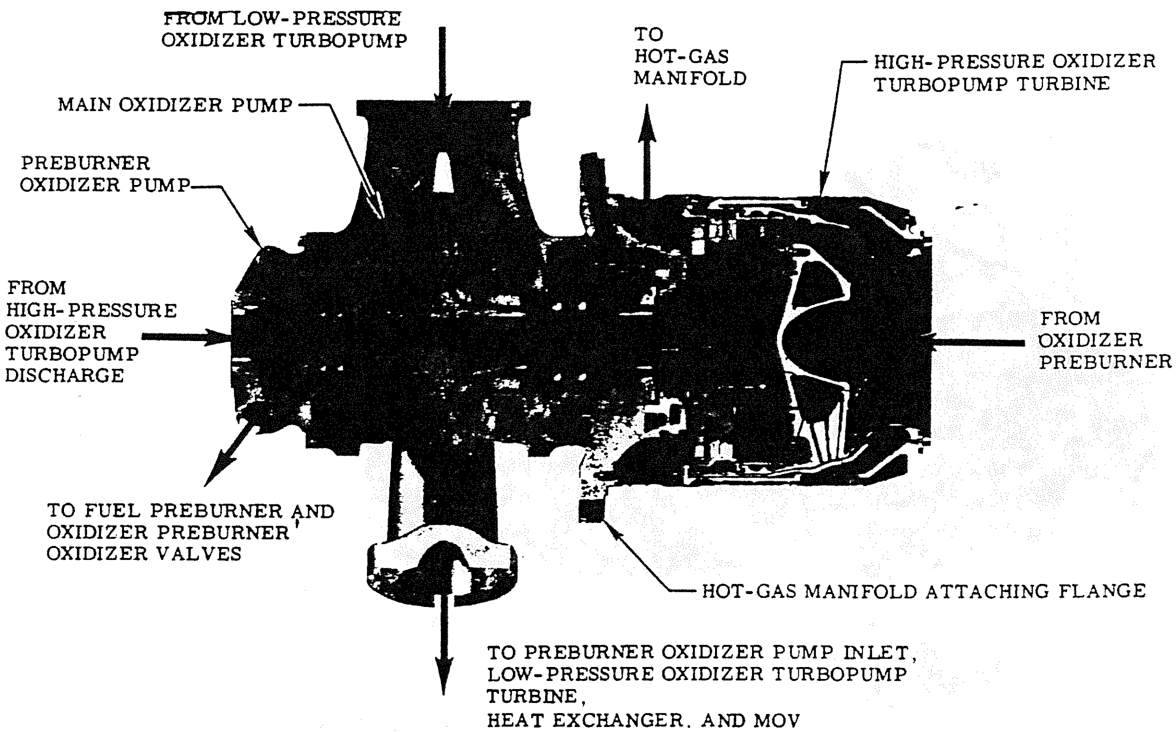


Figure 1.2-11.- High-pressure oxidizer turbopump.

1.2.2.9 High-Pressure Oxidizer Turbopump Turbine

The high-pressure turbopump turbine is powered by hot gas generated by the OPB (fig. 1.2-12). Hot gas enters the turbine and flows across the shielded support struts, through the first- and second-stage nozzles and blades, and is discharged into the HGM. All components are cooled by GH2 flowing over or through them. Coolant is supplied from the OPB coolant jacket. After cooling the turbine components, the coolant is exhausted into the hot-gas flow stream.

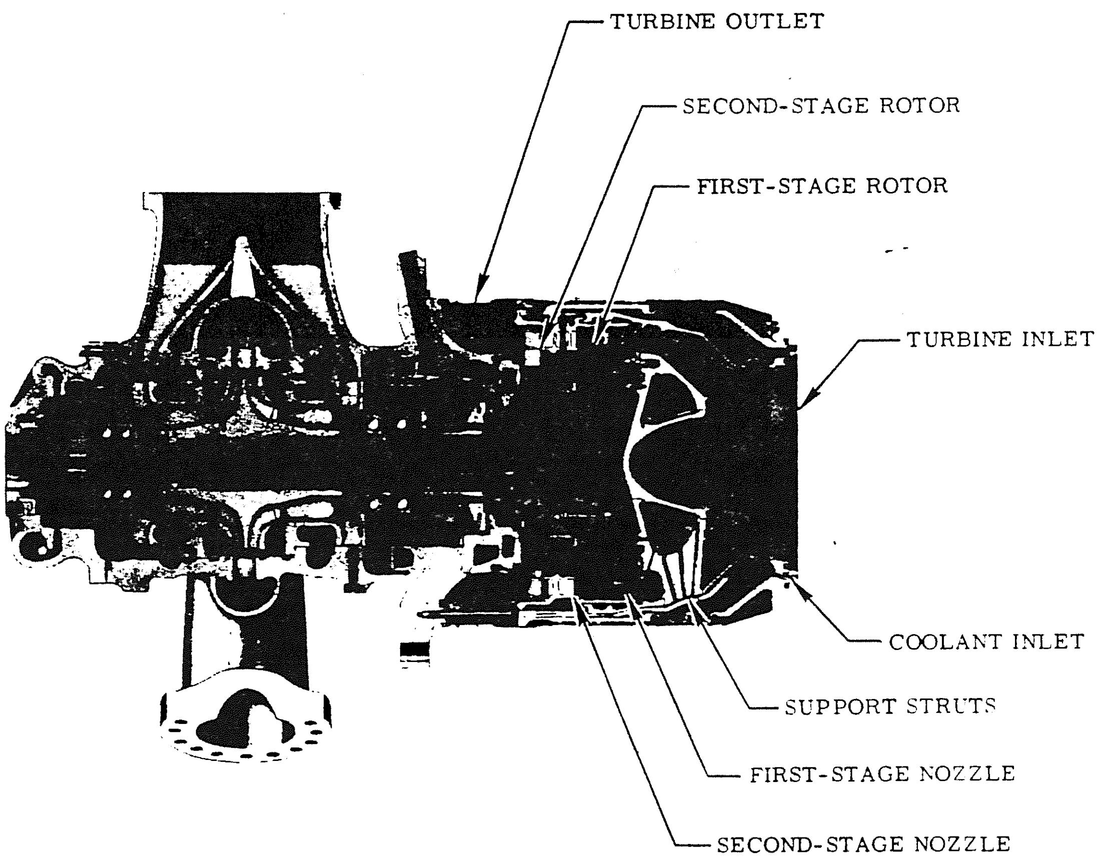


Figure 1.2-12.- High-pressure oxidizer turbopump turbine.

1.2.2.10 High-Pressure Fuel Turbopump

The HPFT is driven by a two-stage hot-gas turbine (fig. 1.2-13). The pump receives fuel from the LPFT and supplies it at increased pressure, through the MFV, to thrust chamber assembly coolant circuits. Fuel flows in series through the three impellers from pump inlet to outlet, with flow redirected between the impellers by interstage diffusers.

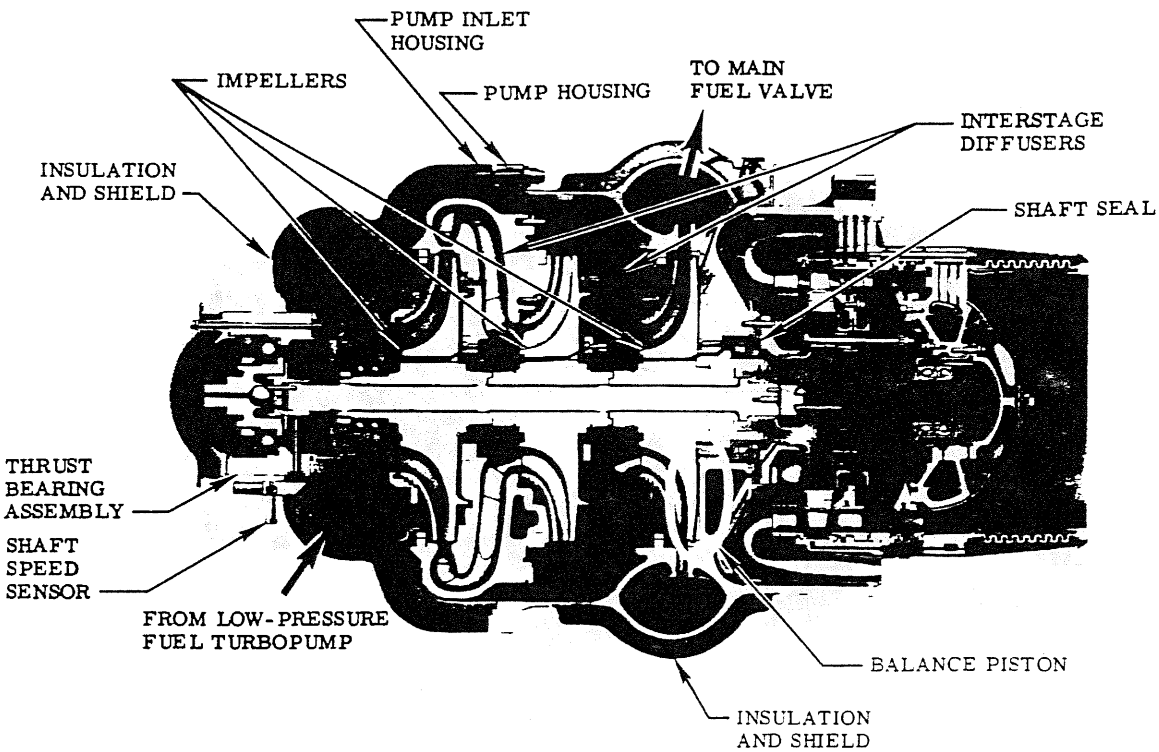


Figure 1.2-13.- High-pressure fuel turbopump.

1.2.2.11 High-Pressure Fuel Turbopump Turbine

The turbine is powered by hot gas generated by the FPB (fig. 1.2-14). Hot gas enters the turbine and flows across the shielded support struts, through the first- and second-stage nozzles and blades, and is discharged into the HGM. The turbine coolant flows over or through the hot gas components and is then exhausted into the hot-gas flow stream.

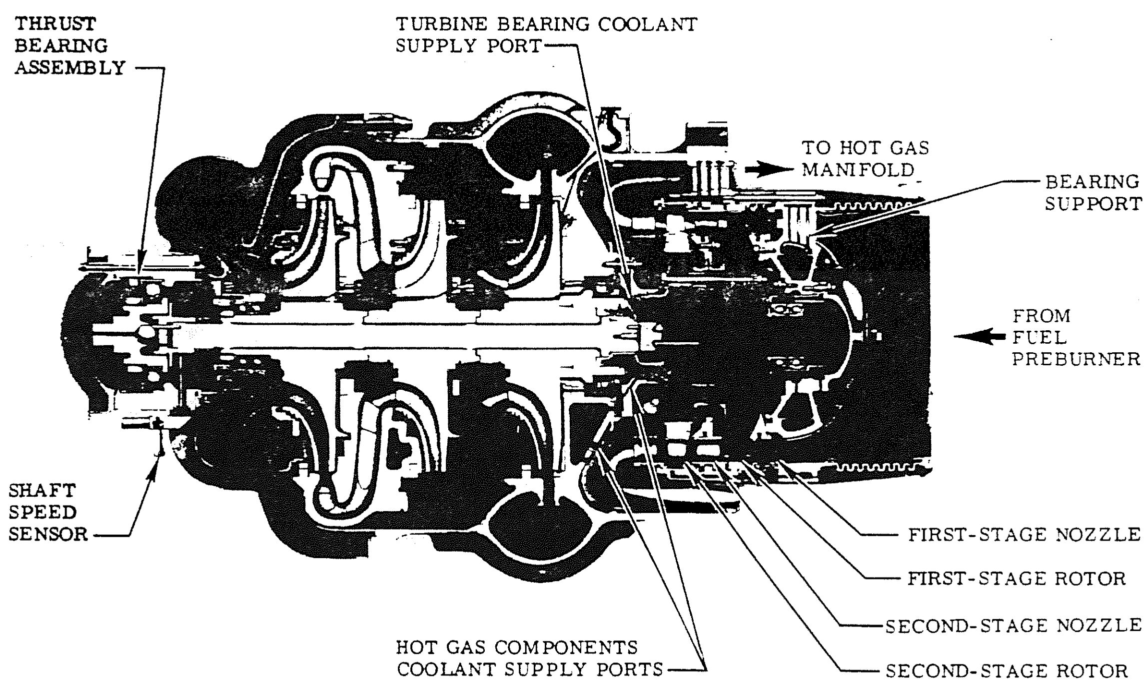


Figure 1.2-14.- High-pressure fuel turbopump turbine.

1.2.2.12 Heat Exchanger

The heat exchanger (HEX) converts liquid oxygen to gaseous oxygen for vehicle oxygen tank pressurization and the POGO system accumulator (fig. 1.2-15). Liquid oxygen, tapped off the discharge side of the HPOT, is supplied to the inlet of the HEX through an AFV. The oxygen is heated to a gas in the small tube (first stage) and nearly to the heating gas temperature in the two larger tubes (second stage). The orificial bypass line injects an unheated portion (approximately 35 percent) of the total oxygen flow into the outlet of the coils for control of temperature/flow-rate operating characteristics. Orifices in the vehicle and POGO system control the HEX oxygen flow rate.

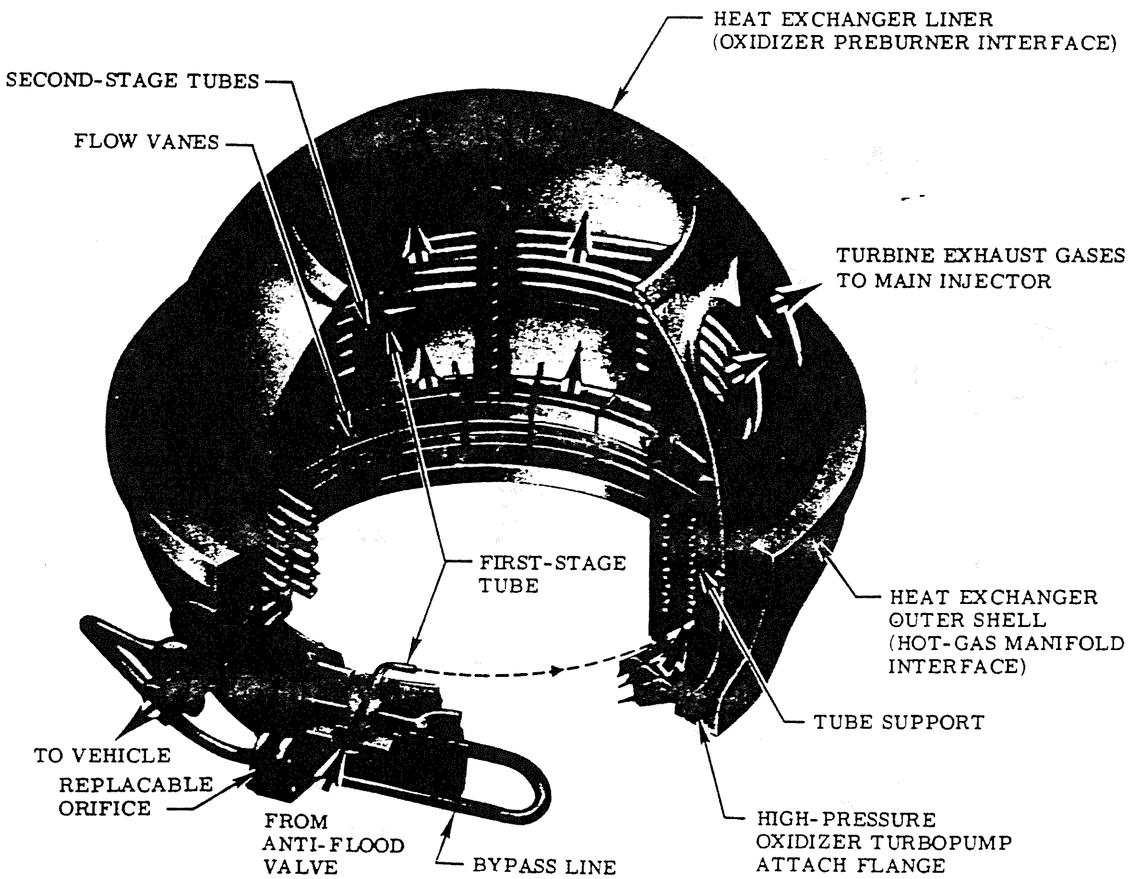


Figure 1.2-15.- Heat exchanger.

1.2.2.13 POGO Suppression System

Flow oscillations transmitted from the Space Shuttle vehicle are suppressed by a gas-filled accumulator which is flange attached to the HPOT inlet duct (fig. 1.2-16). G02 provided to the accumulator from the HEX is used as the compliant medium following an initial helium precharge. A continuous G02 flow is maintained at a rate governed by the engine operating point. Liquid level in the accumulator is controlled by a standpipe in the accumulator, which is constructed with orifices to regulate the G02 overflow throughout the engine operating power level range. The system is sized to provide sufficient G02 to replenish condensation at the minimum G02 flow rate and to permit sufficient G02 overflow at the maximum decreasing pressure transient in the LPOT discharge duct. At all other conditions, excess gaseous and liquid oxygen are recirculated to the LPOT inlet through the engine oxidizer bleed duct.

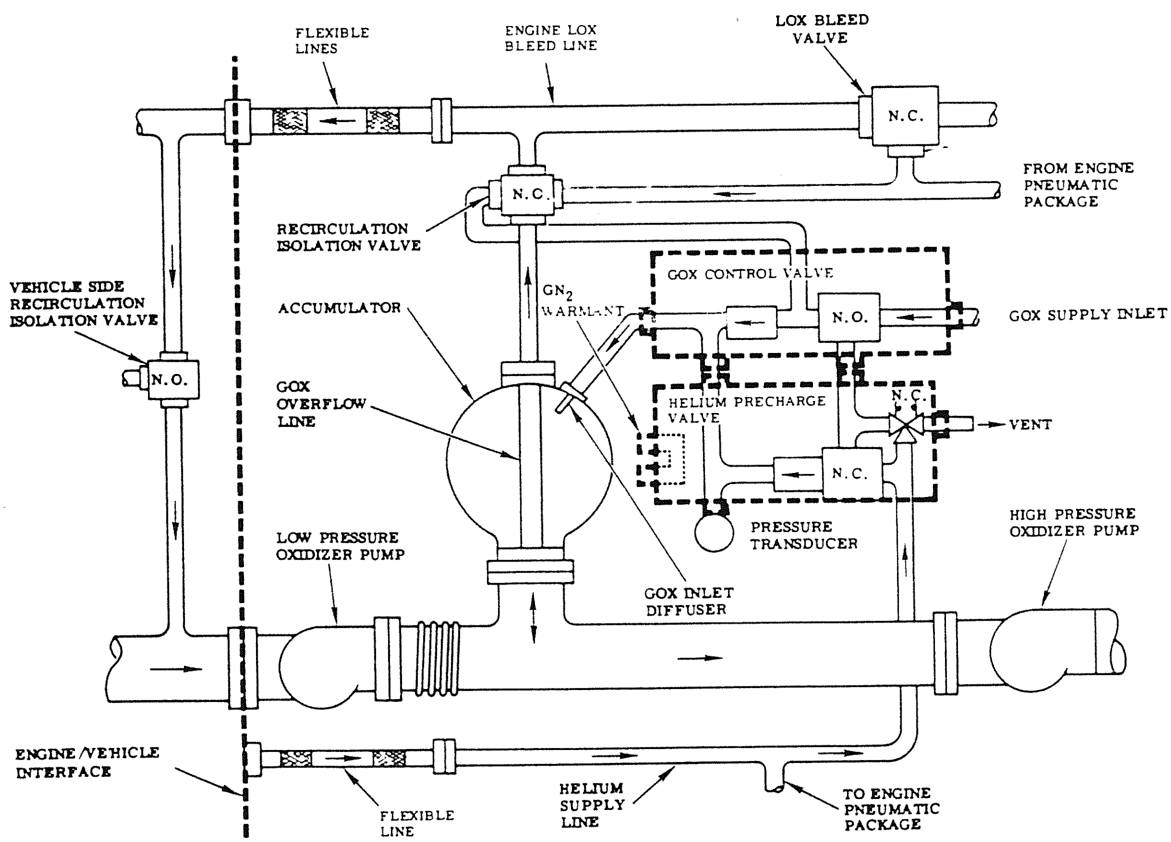


Figure 1.2-16.- Pogo suppression system schematic.

1.2.2.14 Valves (reference fig. 1.2-1)

A. Recirculation Isolation Valve (RIV)

The RIV prevents a short circuit in the engine oxidizer system during the propellant conditioning mode of engine start preparation.

B. Main Oxidizer Valve (MOV)

The valve control oxidizer flow to the main chamber LO2 dome and main chamber ASI.

C. Main Fuel Valve (MFV)

The valve permits or stops the flow of fuel to the thrust chamber coolant circuits, the LPFT turbine, the HGM coolant circuit, the OPB, the FPB, and the three ASI's.

d. Fuel Preburner Oxidizer Valve (FPOV)

The valve permits or stops the flow of oxidizer to the FPB and the FPB ASI.

e. Oxidizer Preburner Oxidizer Valve (OPOV)

The valve permits or stops the flow of oxidizer to the OPB and the OPB ASI.

F. Chamber Coolant Valve (CCV)

The gate-type valve serves as a throttling control to maintain proper fuel flow through the main combustion chamber and nozzle coolant circuits. It is installed in the chamber coolant valve duct.

G. Propellant Bleed Valves

The Oxidizer Bleed Valve (OBV) and the Fuel Bleed Valve (FBV) provide a recirculation flow for propellants through the engine to insure that propellants in the engine are at the required temperatures for engine start.

H. Anti-Flood Valve (AFV)

The valve prevents the flow of liquid oxygen into the heat exchanger until sufficient heat is applied to he HEX during engine start to convert the liquid oxygen to gaseous oxygen.

I. Purge Check Valves (PCV)

The valves isolate propellants from the pneumatic systems.

1.2.2.15 ASI Injector/Combustion Chamber

Orifices comprising the fuel injector direct the ASI fuel into the ASI combustion chamber downstream of the oxidizer orifices (fig. 1.2-17). This injection flow pattern creates an oxidizer-rich condition prior to ignition in the vicinity of the spark igniter electrodes. After ignition, the flow pattern develops into a relatively low mixture ratio environment around the spark igniter electrodes, with an oxidizer-rich zone that protects the ASI combustion chamber walls.

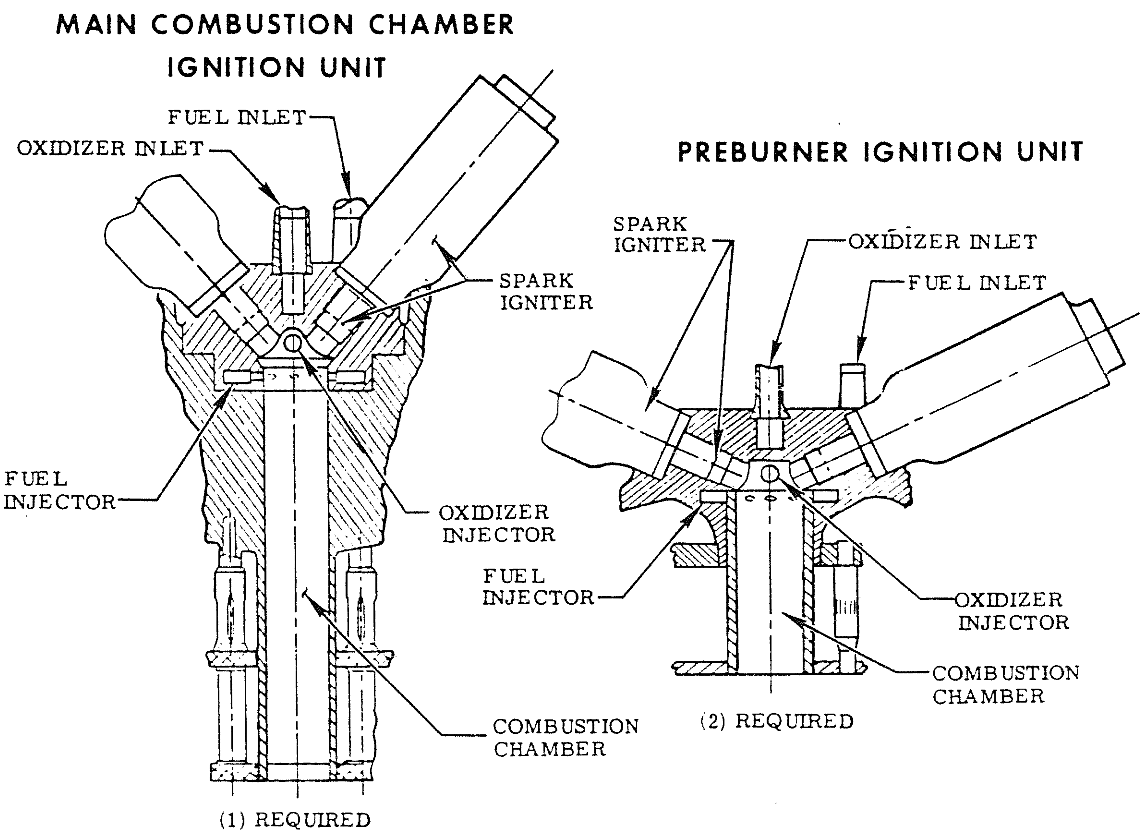


Figure 1.2-17.- The ASI injector/combustion chamber.

1.2.2.16 Interconnect

Fluid interface lines are the vehicle-to-engine lines for recirculation of propellants, propellant tank pressurants, hydraulics, and pneumatics. Fuel ducts and lines in the propellant recirculation (conditioning) system are insulated to prevent liquid air condensation and to help maintain fuel temperature at the desired level.

Main propellant articulating ducts interconnect the nongimbaled low-pressure turbopumps to engine components that are gimbaled. See figure 1.2-18.

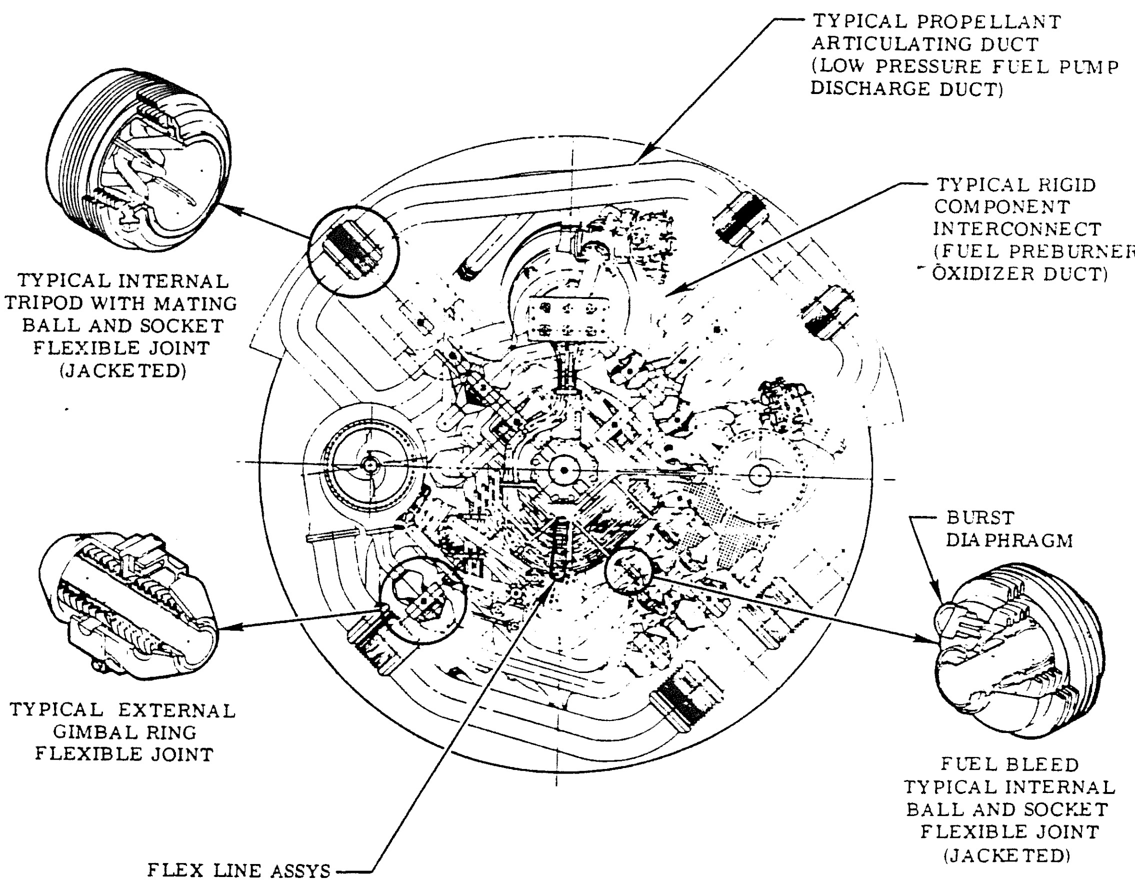


Figure 1.2-18.- Interconnects.

1.3 SSME GUIDANCE CONTROLLED THROTTLING

1.3.1 General

This systems brief describes how guidance software operates to control SSME throttling. Also described are the routines that control open loop thrust bucket commands, adaptive guidance, 3-g throttling, RTLS fuel dissipation, minimum throttling setting for MECO (fine count), and TAL abort auto throttle down. This systems brief explains the THRUST FACTOR that is computed by guidance and is used in the MCC as a cue for an engine-out behind a data path failure.

1.3.2 First-Stage Guidance

The first-stage guidance software is composed of two tasks. The first task sends out four prestored throttle commands to limit the aerodynamic pressure on the vehicle; this is known as the thrust bucket. The second task is a one-time-only calculation that modifies one of the prestored throttle commands based on performance dispersions of the SRB; this is known as adaptive guidance. These two tasks are explained in detail below.

1.3.2.1 Open Loop Thrust Bucket Command

The boost throttling task provides an open loop SSME throttle command to limit the maximum value of dynamic pressure and to gain the desired performance. If an engine fails, this task is not performed and the throttle command is set to the maximum (i.e., 104 percent).

The throttle command (K_CMD) at lift-off is initialized at 100 percent. A table of four velocity break points ($QPOLY_{1,2,3,4}$) and a table of four throttle settings ($THROT_{1,2,3,4}$) are stored in the software. As long as manual throttling is not being performed, when the relative velocity (V_RHO_MAG) equals or exceeds the first stored velocity break point, $QPOLY_1$, the throttle command is changed to the first stored throttle setting ($THROT_1$). The index pointer (J) for these two tables of I-loads is then incremented by one. Each time the relative velocity equals or exceeds the next velocity break point (pointed to by J), K_CMD is changed to the next throttle setting (pointed to by J) and J is then incremented by one. I-loaded values for QPOLY and THROT shape the nominal thrust bucket. See table 1.3-I for typical values. The values are flight specific.

1.3.2.2 Adaptive Guidance

The boost guidance task is performed as long as no engines are out. A one-time-only calculation is made to modify the pitch and throttle commands to reduce the impact of SRB performance dispersions.

TABLE 1.3-I.- I-LOAD TABLES

J $\rightarrow$	$QPOLY_1$ =	60 fps	VREF_ADJUST = 358.4 fps
	$QPOLY_2$ =	400 fps	
	$QPOLY_3$ =	658 fps	
	$QPOLY_4$ =	1280 fps	
J $\rightarrow$	$THROT_1$ =	104 pct	
L_THRT $\rightarrow$	$THROT_2$ =	104 pct	$\longrightarrow$ K_CMD
L_THRTL $\rightarrow$	$THROT_3$ =	67 pct	
	$THROT_4$ =	104 pct	

For low performing SRB's, the throttle level is increased and the vehicle is pitched down. For high performing SRB's, the throttle level is decreased and the vehicle is pitched up. Adaptive guidance can increase payload capability by as much as 1000 lb.

When the relative velocity (VRHO_MAG) reaches an I-loaded reference velocity (VREF_ADJUST), a time difference (TDEL_ADJUST) is calculated by subtracting the mission elapsed time (T_MET) from an I-loaded time value (TREF_ADJUST). If the MET was equal to TREF_ADJUST, the value of TDEL_ADJUST would be zero, and the vehicle had reached the reference velocity as predicted. If TDEL_ADJUST is greater than zero, the vehicle reached the reference velocity later than predicted, and the acceleration was less than expected (cold SRB case). If TDEL_ADJUST is less than zero, the vehicle reached the reference velocity sooner than predicted, and the acceleration was greater than expected (hot SRB case).

Therefore, the sign of TDEL_ADJUST is determined by hot or cold SRB's, and the magnitude of TDEL_ADJUST is proportional to the dispersion from nominal performance. If TDEL_ADJUST equals zero, no throttle settings are updated and the nominal thrust bucket is flown. If TDEL_ADJUST is greater than zero (cold SRB's), the third prestored throttle settings ($THROT_3$ pointed to by LTHRTL index) is increased by a factor of a low performance I-load and the magnitude of TDEL_ADJUST. If TDEL_ADJUST is less than zero (hot SRB's), the the second prestored throttle setting ($THROT_2$ pointed to by LTHRT index) is decreased by a factor of a high performance I-load and the magnitude of TDEL_ADJUST. The modified throttle setting must be between two I-loaded limits. See figure 1.3-1.

The following equations are taken from the Guidance Ascent/RTLS GN&C FSSR, STS 83-0002A (BFS is the same as PASS (ref. PRD MG038104)).

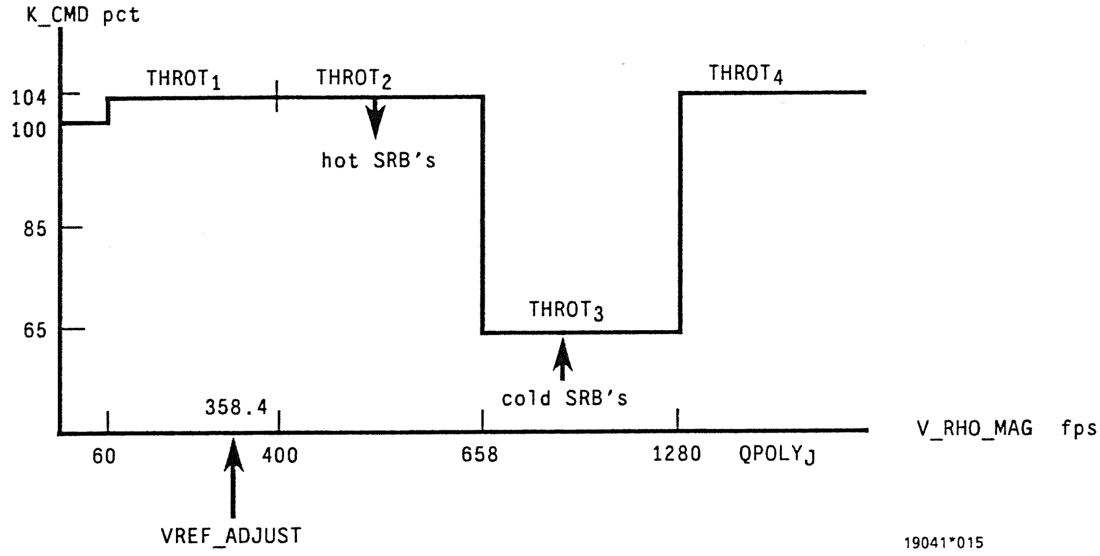


Figure 1.3-1.- Nominal throttle bucket.

If V_RHO_MAG > VREF_ADJUST

TDEL_ADJUST = TMET - TREF_ADJUST

If ABS (TDEL_ADJUST) < TDEL_ADJUST_DEADBAND

TDEL_ADJUST_USE = 0.0 to prevent updates for small dispersions

otherwise TDEL_ADJUST_USE = TDEL_ADJUST

If TDEL_ADJUST_USE > 0, then:

KMIN ALT = 67 percent

THRT_FAC = THRT_FACL

L_THRT = L_THRTL

$$THROT_{L\_THRT} = THROT_{L\_HRT} + ROUND\frac{(THRT\_FAC * TDEL\_ADJUST)}{(QPOLY_{L\_THRT+1} + QPOLY_{L\_THRT})}$$

$$THROT_{L\_THRT} = MIDVAL(KMIN\_ALT, THROT_{L\_THRT}, THROT_4)$$

where:

VRHO MAG	=	relative velocity
TGD	=	current GMT
T_GMTLO	=	GMT of lift-off
TREF_ADJUST	=	I-load time to reach reference velocity (17.28s)
VREF_ADJUST	=	I-load reference velocity (358.4 fps)
TDEL_ADJUST	=	time difference between MET and TREF_ADJUST
KMIN_ALT	=	minimum throttle requirement (altitude constraint)
THRT_FAC	=	I-load throttle adjustment factor for hot SRB's (4000 pct*fps/s)
THRT_FACL	=	I-load throttle adjustment factor for cold SRB's (7500 pct*fps/s)
L_THRT	=	throttle table index for hot SRB's I-load (2)
L_THRTL	=	throttle table index for cold SRB's I-load (3)

1.3.3 Second Stage Guidance

1.3.3.1 Parameter Reinitialization Task

There is software to refresh the throttle command at the RTLS/AOA abort mode boundary time (I-load) as long as no engine has failed, manual throttles have not been selected, and no abort mode has been selected. However, this software has been no-op'ed by making the abort mode boundary time a very large number.

When T_NAV > R_RTLS_AOA + T_GMTLO

K_CMD_MODE_BNDRY

where:

T_RTLS_AOA (V97U4661C) = 1000 seconds
K_CMD_MODE_BNDRY (V97U4420C) = 104 percent

1.3.3.2 TAL Auto Throttle Down

When a TAL abort is selected late, the engines are throttled back to the minimum power level to allow additional time for an OMS dump before MECO. When a TAL abort is selected, the engines are throttled back if the vehicle is above an I-loaded velocity threshold. There are two tables of velocity thresholds, one for the three-engine TAL and one for the two-engine TAL. Each table has specific I-loads for different abort sites.

If either N_SSME = 2 and VGDMAG > TARGET12(TAL_AREA)

or N_SSME = 3 and VGDMAG > TARGET13(TAL_AREA)

then K_CMD = K_MIN (67 percent)

1.3.3.3 3-g Throttling

One part of the second-stage guidance task is the g-limiting task (G LIM TSK), which determines when the 3-g limit has been reached and computes the SSME throttle command (K_CMD) to limit the average acceleration on the vehicle to 3-g's. The throttling algorithm works on an integral and proportional error. This algorithm tends to drive the actual acceleration of the vehicle to the acceleration limit "on the average" since an integral error is included. This algorithm is explained below.

When the current measured vehicle acceleration (AT_AVE) is greater than the premission I-loaded constant (ALIM2 = 3), the throttle setting (K_CMD) is updated, based on a function of the previous throttle setting, the current acceleration (AT_AVE), and the desired acceleration (AD).

K_CMD = ROUND((K_CMD)*(AD)/(AT_AVE))

The current vehicle acceleration (AT_AVE) in the equation above is calculated by dividing the measured delta-v by delta-t.

AT_AVE = DVSMAG/DTGD

where:

AT_AVE = current measured vehicle acceleration
DVSMAG = magnitude of the DVS vector
DTGD = difference between current and previous values of GMT

The desired acceleration (AD) in the above equation is determined from proportional and integral errors. These errors are the difference between the actual acceleration (AT_AVE) and a specific force limit used during SSME maneuvers (AL). The proportional error is calculated from:

DA = AT_AVE - AL

This error (DA) is the integrated over time by the following summation:

SUM_DA = SUM_DA + (DA)*(DTGD)

The desired acceleration of the vehicle (AD) can now be computed by taking the previous desired acceleration and subtracting the current (proportional) acceleration error multiplied by the proportional gain (K_PROP) minus the accumulated (integral) acceleration error multiplied by the integral gain (K_INT). See the following equation.

AD = AD - (K_PROP*DA + K_INT*SUM_DA)

If the elapsed time from the last throttle command (T_KCMD) exceeds the I-load throttle lag time (TLAG), then the following computations are performed.

K_CMD = ROUND ((K_CMD)*(AD)/(AT_AVE))

This equation updates the desired throttle setting as a function of the previous throttle setting, the desired acceleration, and the current acceleration. The throttle setting is rounded off to the nearest percent.

Next, the throttle setting is limited between KMIN and KMAX.

K_CMD = MIDVAL(KMIN, K_CMD, KMAX)

The ratio of AD/AT_AVE can be thought of as the slope of the throttle vs. time plot. If the ratio AD/AT_AVE is less than one, it can be thought of as the throttle down rate. If an engine shuts down during 3-g throttling (such as the manual shutdown at 23K fps), the measured acceleration AT_AVE would drop and the ratio AD/AT_AVE would suddenly increase. When the engine shuts down during 3-g throttling, the other two are throttled up. The throttle command will be limited to be no greater than KMAX (104 percent). Once the other two are throttled up to 104 percent, they are throttled down to maintain 3 g's.

1.3.3.4 Fine Count

The period of time from the initiation of the minimum throttle command and MECO is known as fine count or the MECO PREP TASK. Guidance software commands the minimum throttle setting, calculates the cutoff velocity, and sets the MECO time. The primary task performed by guidance immediately before and during fine count is computing the appropriate MECO time, based on the desired cutoff velocity. When calculating the MECO time, the velocity change during the next guidance cycle, the velocity gained while throttling down from the current throttle setting to the minimum power level, and the velocity gained while the vehicle is at the minimum throttle setting are calculated. The minimum throttle setting must be commanded at a time that ensures that minimum thrust is applied for the necessary time period before MECO. This minimum time is an I-load approximately equal to 6.9 seconds. When MECO is less than two guidance cycles away, a flag is set, signaling the SSME OPS function to initiate the nominal countdown timer. When manual throttles are selected, manual guidance is also selected, and neither the minimum thrust command nor the MECO command are sent from guidance. If the crew has selected manual throttles, they have to perform manual MECO. The detailed requirements of fine count follow.

The additional velocity needed to reach MECO velocity (VGO) is calculated as follows:

VGO_FCD = VDMAG - VMAG

where:

VDMAG is the desired MECO velocity
VMAG = ADVAL(VGD) is the magnitude of the current velocity vector

Whenever this remaining velocity becomes less than a limit, the current throttle command becomes the cutoff (MECO) throttle command.

K_CMD = C_CO

The cutoff throttle setting is chosen according to how many engines are running (N_SSME).

If N_SSME = 3, then K_CO = KMIN (67 percent)
otherwise, K_CO = K_CO_MAX (91 percent)

The VGO limit is updated every guidance cycle. The limit is the sum of three terms: the predicted velocity change over the next guidance cycle, the predicted velocity change during throttle down from the current throttle setting to the cutoff throttle setting, and the velocity change while the SSME's are at the minimum throttle level for the desired I-loaded time.

a. The velocity change in the next guidance cycle is the acceleration multiplied by the delta time

deltav = a*dt = AT_FCD*DT_FCD

where:

AT_FCD is the component of the measured acceleration vector along the current velocity vector, computed by the dot product of the two vectors, and DT_FCD is the planned guidance cycle time for the cutoff task (3.2 seconds).

AT_FCD = (VGD*DVS)/(VMAG DTGD)

DT_FCD = I-load (3.2 seconds)

VMAG = ABVAL(VGD) is the magnitude of the current velocity vector, and DTGD is delta-T between the current and previous state vector times.

b. The velocity change during the time required to reduce the throttle from the current setting to the minimum setting is predicted with the following equation:

$$\text{DV\_RAMP} = (\text{KCMD} + \text{K\_CO})*(\text{K\_CMD} - \text{K\_CO})*\text{AT\_FCD}/(2\text{K\_CMD}*\text{K\_DOT})$$

c. The velocity change during the desired time at minimum throttle setting is the deltav times the ratio of the cutoff throttle command to the current command

$$\text{DV\_MIN\_K} = \text{AT\_FCD}*\text{DT\_MIN\_K}*\text{K\_CO}/\text{K\_CMD}$$

where:

DT_MIN_K is an I-load (6.9 seconds)

When VGO is less than the limit, the minimum throttle command is sent.

If $\text{VGO\_\_FCD} < (\text{AT\_FCD}*\text{DT\_FCD} + \text{DV\_RAMP} + \text{DV\_MIN\_K})$

then:

$$\text{K\_CMD} = \text{K\_CO}$$

and:

S_KCO = ON to indicate that pre-MECO throttle down is complete

The time left until MECO (TGO) is determined by dividing the velocity left to go by the acceleration and then subtracting a time factor due to tailoff.

$$\text{TGO} = \text{VGO\_FCD}/\text{AT\_FCD} - \text{DT\_TAILOFF}_{\text{ITLOF}}$$

where:

the index ITLOF = MAX(INTEGER(N_SSME-1,1))

When an RTLS abort is chosen, the time to go is calculated by dividing the velocity to be gained by its rate of change:

$$\text{TGO} = \text{VGOMAG}/\text{VGO\_DOT}$$

For either a nominal or an RTLS case, when time to go becomes less than a specified number of guidance cycles, the time to command MECO becomes T_MECO = TGD + TGO and a flag is set to initiate countdown (S_TMECO). The required MECO accuracy is ±40 ms. This variation, the difference between the desired start to tailoff and the actual start of thrust tailoff, is a function of GPC software timing and engine controller timing variation.

1.3.3.5 RTLS

The RTLS uses the same general purpose tasks as second stage guidance to limit the vehicle to 3-g's and to command minimum throttles for fine count. This systems brief describes three major phases of an RTLS abort: fuel dissipation, flyback, and powered pitchdown.

1.3.3.6 RTLS Initialization

The following calculations are performed only once on the transition to MM601, a one-time-only calculation made to set the current/fuel dissipation (K_CMD) throttle setting and the flyback throttle setting (K).

If N_SSME = 3

K_CMD = MAX(ROUND(2*KMAX/3),KMIN) = MAX(2*104/3,67) = 69

and

K = K_3ENG (72 percent)

If an engine has failed, the engines are not throttled down and the fuel dissipation throttle setting is 104 percent. Only the flyback throttle setting is calculated.

If N_SSME < 3

K = K_2ENG

1.3.3.7 Fuel Dissipation

There are five tasks performed during RTLS fuel dissipation: setting the flyback and pitch command discretes, changing the roll command, calculating the turnaround time, calculating the predicted MECO time, and, if manual guidance is selected, setting the guidance discretes to their proper values.

1.3.3.8 Max Throttles

For any engine fail in first stage (N_SSME < 3), any time in second stage, or any time during a contingency abort when max throttles are selected, max throttles (109 percent) are commanded.

If MAX_THROT_CMD = ON and N_SSME = 3

then:

K_CMD = K_MAX_SECONDARY (109 percent)

For an engine out RTLS, when max throttles are selected, max throttles are commanded.

If MAX_THROT_CMD = ON and N_SSME = 3

then:

K_CMD = K_MAX_SECONDARY (109 percent)

1.3.3.9 RTLS Flyback

There are seven subtasks performed by RTLS flyback: obtain an estimate of filtered acceleration, exhaust velocity, and mass; extrapolate acceleration, position, and time to the end of the turn; execute PEG calculations and maintain convergence; calculate the estimated time of MECO; issue throttle commands; set the fine countdown discrete on when the time-to-go (TGO) is less than a K-load limit; and, if manual guidance is selected, set the proper discretes. At the end of the turnaround, after guidance has been converged and the difference between the current throttle setting (K_CMD) and the PEG throttle (K) is greater than a K-loaded threshold (0.75), the current throttle command is set to the PEG throttle command.

If ABS(K_CMD - K) > DK_ROUND

then:

K_CMD = ROUND(K)

During RTLS flyback, like the fuel dissipation phase, if max throttles are selected, only the KMAX I-load is updated from 104 to 109 percent for any number of engines running. Guidance does not command 109 percent until it needs it.

KMAX = KMAX_SECONDARY

1.3.3.10 RTLS Powered Pitchdown

There are four subtasks of powered pitchdown: compute the quaternion for the powered pitchdown, mated coast, and -Z translation; command the engines to the minimum throttle setting; set the powered pitchdown discrete on; and delay the completion of powered pitchdown for late engine failures.

If manual guidance and manual throttling are not being performed, the engines are commanded to the minimum setting.

K_CMD = KMIN (67 percent)

1.3.4 Powered Contingency Abort Sequencing

The crew may manually initiate automatic contingency abort sequencing software (available in MM103 and MM601) via item entries for arm and start on the TRAJ displays. The TRAJ displays show the current contingency abort procedure region (BLUE, GREEN, etc.) that corresponds to the contingency abort cue cards. The abort regions are determined by each guidance cycle by comparing vehicle inertial velocity (VI_MAG), altitude rate (HDOT), and equivalent airspeed (EAS) to prestored I-loads. The abort mode colors are displayed to assist the crew in determining their current contingency region status if a second or third engine were to fail at the current time. For nominal, ATO, or TAL, the CONTINGENCY ABORT cue card is used. For an RTLS, the RTLS CONTINGENCY cue card is used.

1.3.4.1 RTLS 2 Out Red

Automatic initiation of the contingency abort sequencing only occurs prior to powered pitchdown and fine count for the second engine fail RTLS case when single engine completion capability does not exist. This is known as the 2 Engine Out RED RTLS CONTINGENCY abort mode 5 task. This task commands a pitch attitude to reach an I-loaded altitude rate before powered pitchdown without exceeding a specified maximum pitch angle. When the time to MECO (TGO) becomes as small as the time required to pitch down, this task commands powered pitchdown to achieve a desired altitude before MECO.

This task also calculates the time to go to MPS propellant depletion (TGO_EMPTY), and if it is less than a I-loaded value, the engines will be throttled down to protect for possible LO_2 NPSP violation if a low level cutoff occurs. This task also predicts the MECO time for display. The details of this task are described below.

To accurately track the vehicle mass, the amount of mass lost due to single engine roll control OMS usage is estimated. On the first pass, the mass value is initialized to zero.

M_SEROLL = 0

On subsequent passes, the mass loss is accumulated at a constant rate.

M_SEROLL = M_SEROLL + DELTA_M_SEROLL

Using this value, the current vehicle mass (M), the total vehicle mass flowrate (MDOT) from other guidance tasks, and an I-load value set to the vehicle mass at low level cutoff (M_EMPTY), this task calculates the time remaining before MPS propellant depletion.

TGO_EMPTY = (M - M_EMPTY - M_SEROLL)/MDOT

This task also calculates the time remaining before a guided MECO by using parameters from the RTLS TRAJ display processor. This task calculates TGO by estimating how long before the display bug reaches the cutoff tic mark.

TGO = -DELTA_R/DELTA_R_DOT

Next, this task compares TGO_EMPTY and TGO and selects the smallest time to MECO.

If TGO_EMPTY < TGO, then set TGO = TGO_EMPTY

TGO, the time to MECO (guided or low-level cutoff, whichever is smallest) is then reset to the time to go before powered pitchdown. The time of powered pitchdown is also computed.

TGO = TGO - DT_CONT_PITCH
TP = TRGD + TGO

where:

DT_CONT_PITCH is the computed time required to complete the pitchdown maneuver and TGD is the current time

Next, the time to go before low-level cutoff and the time to go before the powered pitchdown maneuver are compared to I-load values. If the time to go before low-level cutoff is less than an I-loaded value, the engines are throttled down and the time to complete the pitchdown maneuver is set to an I-loaded value. Otherwise, if the time to go before powered pitchdown is less than an I-load, the fine countdown discrete is set on.

If TGO_EMPTY < TGO_CONT_THROT (V97U8400C = 24 sec)

then:

the engines are throttled down, the time to complete the pitchdown maneuver is set to an I-load time, and fine count is commanded

K_CMD = K_CONT_THROT (V97U8375C = 91 percent)

DT_CONT_PITCH = DT_CONT_THROT (V97U8350C = 13 seconds)

S_CONT_MECO = ON

else:

If TGO < TGO_PPD (K-load = 6 sec) set fine count

S_CONT_MECO = ON

1.3.5 Thrust Factor

1.3.5.1 Background

Part of the second-stage guidance software is the genral purpose acceleration-mass update task (ACC_MASS UPD TASK) whose purpose is to porvide other guidance and targeting tasks with estimates of the current vehicle mass (see SB 2.5) and a scaling factor (FT_FACTOR). FT_FACTOR is used to scale the estimated thrust force on the vehicle so that a smoothed value of the thrust acceleration can be calculated. This scaling factor does not directly influence the throttle commmands to the engine but it is mentioned because it is displayed on the BOOSTER ME display MSK 1051 and MSK 1052. The following paragraphs explain how FT_FACTOR is calcualted and how it is used on console.

1.3.5.2 Part I

FT_FACTOR is an average of the current value of the factor (X1) and the three previous values of the factor (X2, X3, and X4). The current factor is a ratio of two forces, the current vehicle thrust (FT_S), and the predicted thrust (FT). The current vehicle thrust is simply Newton's equation F = ma or F = mdv/dt where dv equals the current vehicle delta-v (DVSMAG) and dt equals the delta-t or difference in the current and previous values of TGD.

All of the equations in this paragraph are shown below.

Update sample factors as shown:

X2 = X1
X3 = X2
X3 = X4

Calculate the current vehicle thrust:

FT_S = M*DVSMAG/DTGD

Where mass (M) is calculated from an approximate exponential extrapolation of sensed velocity (DVSMAG) divided by exhaust velocity (VEX). See SB 2.5.

Calculate the current value of the thrust factor as the ratios of two forces:

X1 = FT_S/FT

Calculate the weighted average of the thrust factor:

FT_FACTOR = (X1 + X2 + 3 + X4) *0.25

1.3.5.3 Part II

All of the variables needed to calculate the thrust factor (FT_FACTOR) are shown above except for the predicated thrust on the vehicle (FT), which is calculated in the thrust parameters task (THRST PRM TSK). The main engine thrust force is calculatd as a function of the current throttle command (K_CMD), the number of main engines thrusting (N_SSME), and an I-loaded estimate of thrust force for a single main engine (FT_SSME). The corresponding estimates of force for the OMS engines and RCS jets are added. See the equation below.

FT = 0.01*(K_CMD)*(N_SSME)*(FT_SSME) + (N_OMS)*(OMS) + (N_RCS)*(FT_RCS)

The number of main engines thrusting (N_SSME) comes from the PFG input task (PFG-INP-TSK) and is determined by subtracting the total number of engine-out flats from three as seen below. The engine-out flags come from the SSME OPS sequence.

Use estimation of mass (Part I)

Estimate of exhaust velocity is estimated thrust over estimate flow rate (MDOT)

VEX = FT/MDOT

where:

MDOT is the sum of estimated mass flow rates from SSME's, OMS engines, and RCS jets

MDOT = 0.01*(KCMD)*(NSSME)*(MDOT_SSME) + (N_OMS)*(MDOT_OMS) + (NRCS + NRCS_NULL)*(MDOT_RCS)

1.3.5.4 Console Use

The nominal value of THRUST FACTOR is 1.0, but if an engine goes out and guidance does not know it (because of a data path failure), the predicted thrust will be wrong. N_SSME will be wrong because the engine-out flags could not be set with a data path failure. The value of THRUST FACTOR will drop by one third of the nominal value and 0.67 will be displayed on main engine display. THRUST FACTOR is not an engine-out cue. If an engine goes out with no other failures, guidance updates its thrust prediction to account for only two engines running (N_SSME = 2) and THRUST FACTOR still has the nominal value of 1.0. THRUST FACTOR is really a cue that tells us that guidance has not been moded after an engine goes out. It is computed only in second-stage guidance only.

If three engines are running and one is stuck in the bucket (67 percent), the actual thrust will be lower than the predicted thrust, and the value of thrust factor would be 0.89. THRUST FACTOR can be used to determine the power level of an engine with a data path failure. Table 1.3-I lists the value of THRUST FACTOR when the commanded power level is 104 percent and one engine is running at an off-nominal power level. The formula can be used when the commanded power level is not 104 percent.

DATA_PATH_PWR_LVL = (3*CMD_PWR_LVL*THRUST_FAC) - (2*CMD_PWR_LVL)

1.3.6 References

Guidance Ascent/RTLS GN&C FSSR

TABLE 1.3-II

CMD PWR LVL = 104 Percent

Data path/stuck throttle PWR LVL	Thrust factor
109	1.016
108	1.013
107	1.010
106	1.006
105	1.003
104	1.000
103	0.997
102	0.994
101	0.990
100	0.987
99	0.984
98	0.981
97	0.978
96	0.974
95	0.971
94	0.968
93	0.965
92	0.962
91	0.958
90	0.955
89	0.952
88	0.949
87	0.946
86	0.942
85	0.939
84	0.936
83	0.933
82	0.929
81	0.926
80	0.923
79	0.920
78	0.917
77	0.913
76	0.910
75	0.907
74	0.904
73	0.901
72	0.897
71	0.894
70	0.891
69	0.888
68	0.885
67	0.881
66	0.878
65	0.875
0	0.667

1.4 TURBOMACHINERY FUNDAMENTALS

1.4.1 General

Turbomachines (including turbines, fans, compressors and pumps) are used to impart/extract work to/from fluids (liquids and gasses). Work here includes the movement and pressurization/depressurization of fluids. For liquid rocket engines, the combination of a turbine and its associated pump is typically called a turbopump.

In liquid rocketry, propellant is typically stored in a tank at low pressure because of the desire for a lightweight tank (lower weight means higher performance). Turbopumps are used to pump propellant from low storage pressure to the required high pressure of the thrust chamber. Because a major goal of a rocket engine is to minimize inert weight and ultimately increase payload, turbopump systems for liquid rocket engines typically have the highest power-to-weight ratio in the field of rotating machinery.

1.4.2 Axial-Flow and Centrifugal-Flow Turbomachines

Two fundamental styles of turbomachines exist: axial-flow and centrifugal-flow. In axial-flow turbomachines, fluid flows approximately parallel to the axis of the rotating shaft. In centrifugal-flow turbomachines, (specifically for a pump or compressor) fluid first flows parallel to the shaft then is turned perpendicular to the shaft. For a centrifugal-flow turbine, flow first enters perpendicular to the shaft and then is turned parallel to the shaft. A combination of axial and centrifugal-flow turbomachines typically is found in a liquid rocket engine.

Historically speaking, the earliest type of axial-flow turbomachine was the Archimedean screw-type pump (fig. 1.4-1). Archimedean screws were found in widespread use circa 200 AD in removing water from underground mines. On rocket engines, this type of pump is now known as an inducer (fig. 1.4-2) and is used in the two low-pressure pumps of the SSME and as part of the high-pressure oxygen pump on the SSME. The screw is a single piece of machined metal. A second type of axial-flow turbomachine of more modern heritage is composed of alternate rows of rotor blades and stator blades (fig. 4.1-3). Rotor blades are airfoils on the periphery of a disk which rotate at high speed. Stator blades (also known as vanes) are located between consecutive rows of rotor blades and are fixed within the housing that contains the turbomachine. Each rotor row and its associated stator row is called a stage (thus named a staged axial-flow turbomachine). The flowing fluid moves through the annular passageway where the blades are located. In staged axial-flow turbomachines, the overall pressure rise/drop is achieved by the summation of pressure rises/drops imparted as the working fluid traverses each stage. Staged axial-flow turbomachines are low in weight and highly efficient. On liquid rockets, staged axial-flow pumps have occasionally been found useful (but not on the SSME) for pumping liquid hydrogen - because of the many stages required. A major problem with

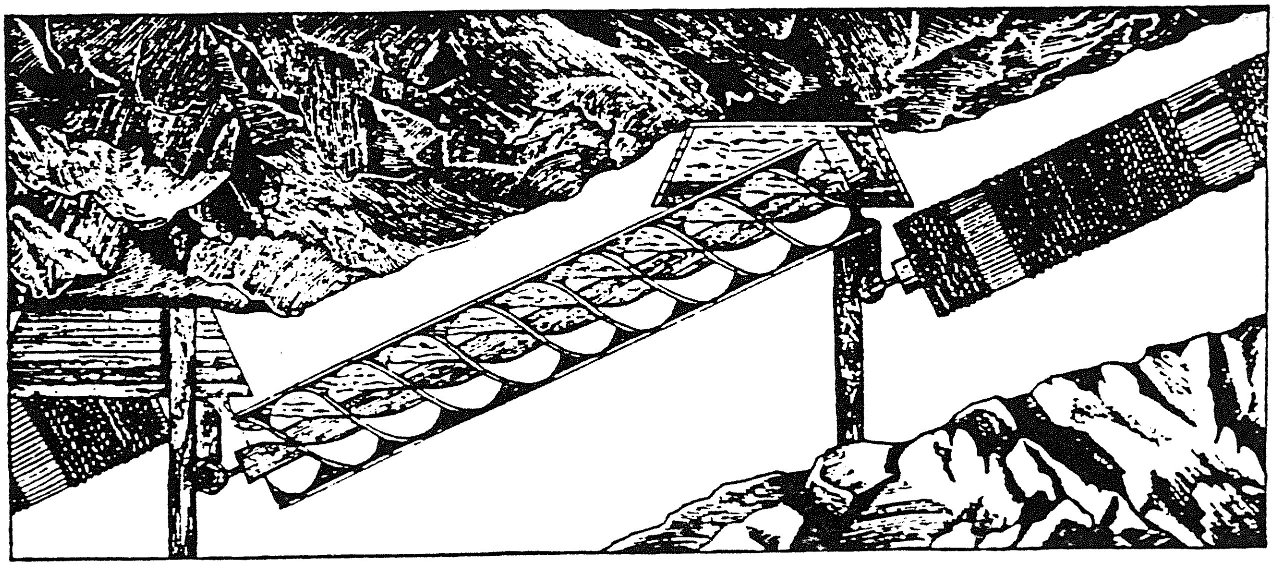


Figure 1.4-1.- Axial-flow turbomachine: Archimedean screw-type pump.

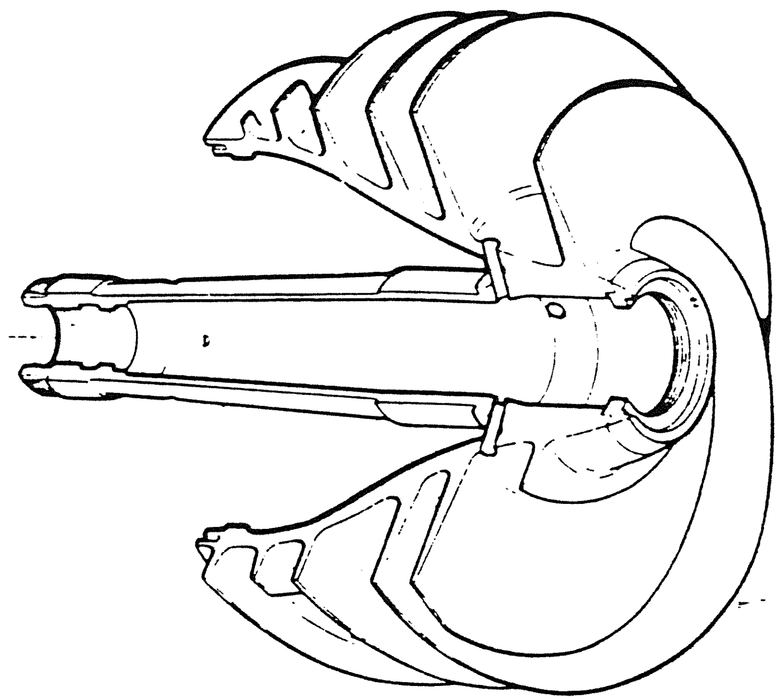


Figure 1.4-2.-Axial-flow turbomachine: Inducer-type pump.

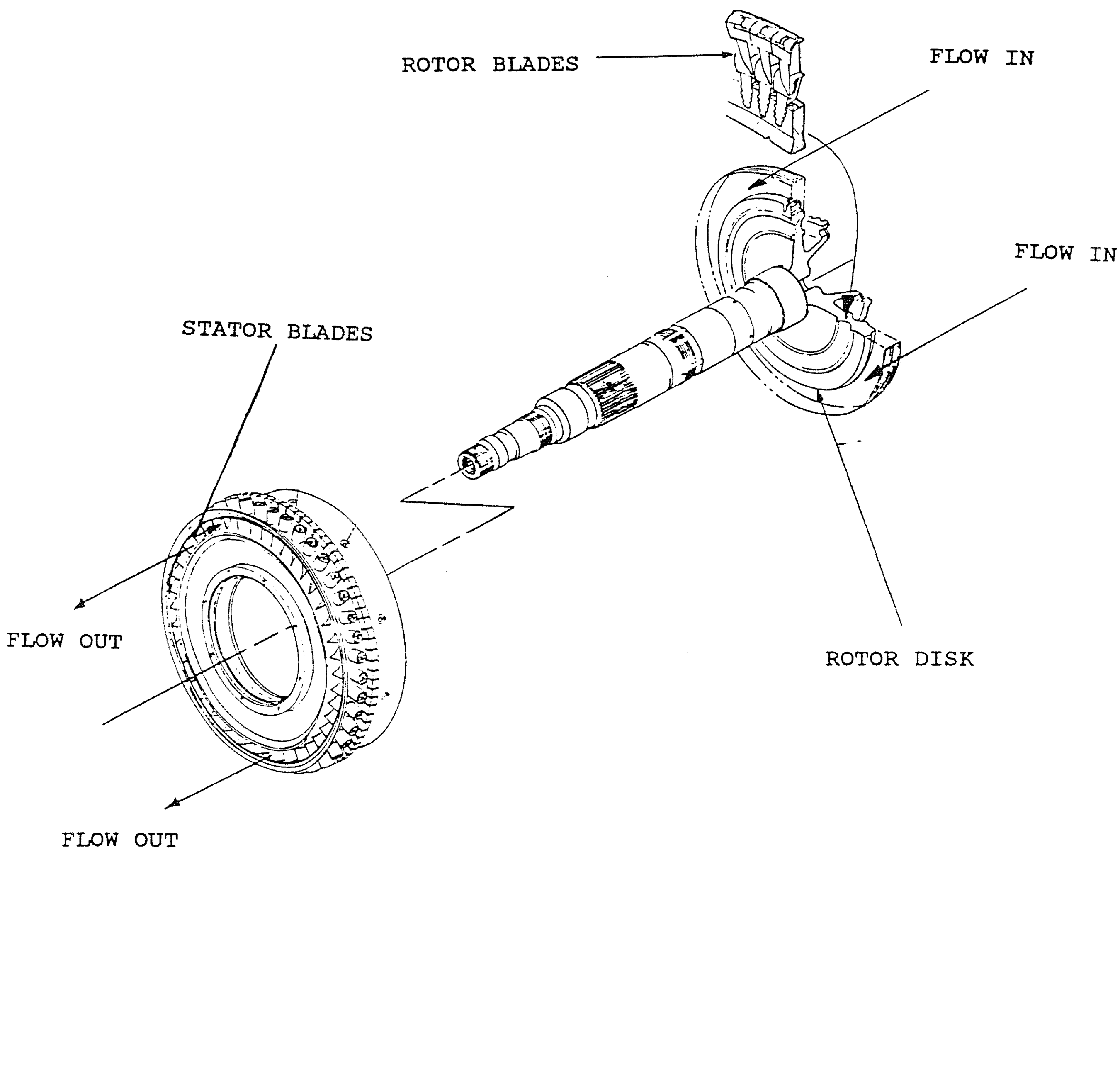


Figure 1.4-3.- Axial-flow turbomachine: Axial-flow-turbine.

staged axial-flow pumps is their usually short service life and small throttling range (about half that of centrifugal pumps). Much more common is the staged axial-flow turbines used to drive most any style of pump. The SSME uses staged axial-flow turbines to drive each of its four pumps.

Centrifugal-flow turbomachines are composed of a bladed rotor (also known as an impeller) shaped such that the rotation (in the case of a pump or compressor) moves the motion of the working fluid from parallel to the shaft to perpendicular to the shaft (fig. 1.4-4). A high-speed and typically spiral velocity is imparted to the fluid. A diffuser, which surrounds the impeller and collects the fluid, converts most of the imparted velocity to a pressure increase (or headrise). The success of centrifugal-flow pumps is due to their simplicity, reliability, light weight, wide operating speed range (throttle ability), short development time, low parts count, and relatively low cost. Other types of pumps become competitive when multi-staging is necessary or maximum efficiency is paramount. Centrifugal-flow turbines basically work just opposite to the above and are not commonly found on liquid rocket engines - they are more typical in automotive applications (turbochargers).

In the SSME, high-speed, staged axial-flow turbines directly drive each pump; this is unlike many other previous rocket engines which have used heavy gearboxes to connect a high speed turbine to one or more lower speed pumps. Turbines for liquid rocket engines differ from other turbines (such as for aircraft engines or steam-electric power generation). Liquid rocket engine turbines operate in an extremely severe environment and thus have short lives. The size and weight are strictly limited. These turbines have a very high specific work output; the fluids contain great amounts of energy. Because of the short starting times and heavy output demands, liquid rocket engine turbines operate under severe thermal shock conditions, high stresses and high loads per stage. Attaining the predicted turbine performance is critical since there is a close interdependence among the efficiency of the turbine, the engine thrust chamber pressure, and the specific impulse of the engine. Typical problems with turbines include failure of blades during operation and failure of forged rotor disks. Also, changes to the pressure ratio can affect the gas-path energy and thus require redesign of the blades. Differential expansion and cracking at the weld joints can require modification of casting and manifold design. Because the turbine provides the power to drive the pump, solving these problems is essential to the overall design of the engine.

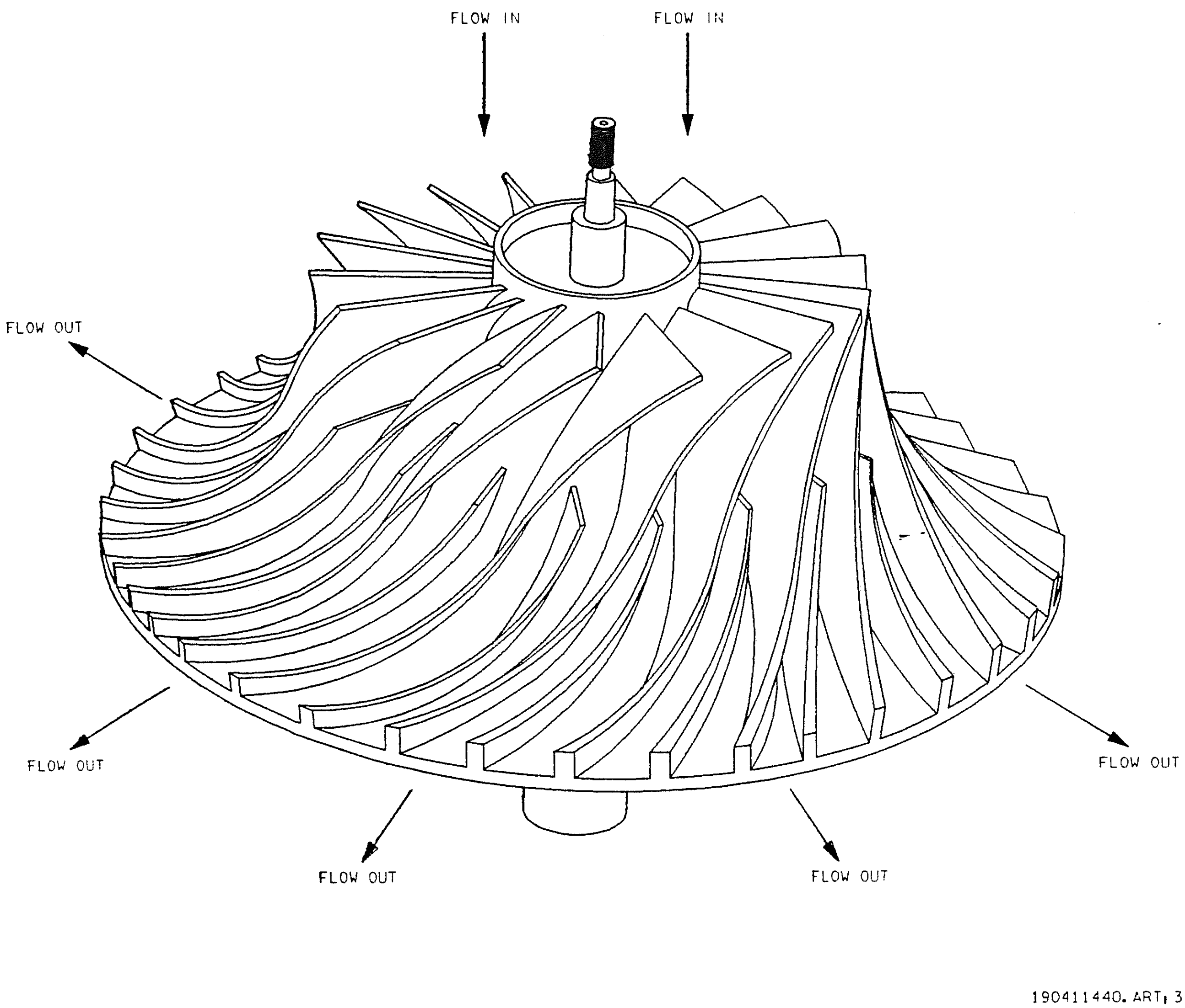


Figure 1.4-4.- Centrifugal-flow turbomachine: Centrifugal compressor.

1.4.3 Design Elements of Turbopumps

General/NPSP

The primary requirement of liquid rocket engine turbopumps is to pump the propellant from the low storage pressure to the required combustion chamber pressure. The use of both low and high pressure pumps on the SSME permits the low storage pressure, which is important because it allows the propellants to be saturated and requires a lighter weight tank. The low storage pressure causes problems with the net positive suction pressure (NPSP) at the inlet. The NPSP is defined as the difference between the fluid pressure and the vapor pressure. If NPSP at a pump inlet is too low, formation of vapor bubbles in the fluid occurs (cavitation) causing mechanical damage to the pump. The pump will no longer deliver its designed pressure rise or may catastrophically self destruct (cavitation unloads the pump, leading to overspeed and turbine disk rupture). Therefore, many pumps are designed for a critical NPSP which is about 2 percent above the value at which cavitation will occur. There are three design corrections for low NPSP. Raising the tank pressure increases the NPSP but with it the tank weight. Decreasing the pump speed also raises the NPSP but lowers the pressure rise. Increasing the inlet diameter (and thus lowering the flow coefficient) increases the NPSP at the expense of lowering the efficiency. A workable solution is the addition of an inducer at the pump inlet. An inducer boosts the pressure of the working fluid before it enters the pump, thus the pump may operate at a lower NPSP. Inducers are mechanically less susceptible to low NPSP cavitation problems so they can be safely used in a low NPSP environment.

a. Headrise/Volumetric Flow Rate

Essential to achieving the desired pump performance is providing the specified head (pressure/density) and quantity of propellant (volumetric flowrate). The pump-developed head is defined as the difference between the discharge head and the suction head. The discharge head must overcome the hydraulic resistances in the system and deliver the fluid to the thrust chamber at the designed chamber pressure. Headrise is commonly approximated by the quotient of the pressure rise and the average propellant density:

$$\Delta H = \frac{\Delta P}{\rho AVG} \qquad \text{Eq 1-4.1}$$

But, more exactly, headrise is defined as the isentropic enthalpy rise for the given pressure rise:

$$\Delta H = \frac{\Delta h_{isen}}{\Delta P} \qquad \text{Eq 1-4.2}$$

For liquid hydrogen pumps especially, effects of compressible flow must be considered in equation 1-4.1 in order to approximate equation 1-4.2.

Propellant heating (due to inefficiency) at one point in the pump can significantly (and non-symmetrically) lower the flow density at a downstream point. Thus average density cannot be used and very complex and expensive design methods (computational fluid dynamics coupled with extensive testing) must be used in the development of advanced turbomachines.

b. Efficiency

To maximize the power-to-weight ratio, an efficient turbopump is needed. The efficiency of a turbopump is the product of the efficiencies of the pump and the turbine. For staged combustion cycles, an increase in turbopump efficiency means an increase in combustion chamber pressure for a given pump input energy. As the pump efficiency increases, the size and weight of the thrust chamber and preburners decrease. Increasing the operating speed allows decreasing the pump and turbine diameters and length and hence the weight. Lightweight pumps can accommodate more payload; small pumps are easier to handle and mount.

For centrifugal pumps, the efficiency increases with increasing speed until a maximum value is reached. Thereafter, the efficiency decreases as speed increases. For axial pumps, the efficiency increases with speed when below a value determined by geometry. Above this value, the efficiency shows little change with speed. Thus, for lower speeds, axial and centrifugal pumps follow the same trend, but at higher speeds, the efficiency of the axial pump levels off while the centrifugal pump decreases.

c. Speed

High pump speeds enhance performance. With high rotational speed comes smaller pressure rise excursions - thus a greater throttling range. When high speeds are used, fewer stages are required to achieve a given pressure rise. The weight of a high-speed turbopump is lower, and the pump and turbine efficiencies are greater. High speeds present many design constraints, however. There are a number of "speed limits" encountered in pump design - some of which are discussed below.

d. Cavitation Speed Limit

The first speed limit is cavitation. The suction speed is inversely proportional to the NPSP: if the inducer operates above the suction speed limit, the NPSP will be too low and the pump will cavitate. When it cavitates, the pump will fail to deliver the designed pressure rise. Hence the pump must be designed to deliver the desired pressure without cavitating. Thermodynamic suppression pressure (TSP) can allow operation near the suction speed limit. TSP occurs when, close to the limit of suction performance, vaporization within the inducer chills the liquid, reducing the vapor pressure and allowing the pump to operate satisfactorily at a lower NPSP. Pre-inducers which boost the NPSP are used to increase the cavitation speed limit.

There are a number of factors involved in the design of turbopump inducers. The inducer is the axial inlet portion of the turbopump rotor. It can be an integral part of the rotor or separate and upstream of the impeller. The function of the inducer is to raise the inlet pressure enough to preclude cavitation in the next stage. Thus the design objective is to achieve high suction performance while retaining structural integrity under the operating conditions. A math model is used to find the highest specific suction speed that will not cause hydrodynamic problems or mechanical problems involving the integrity of the leading edge or blade and build stresses. Cavitation is always a design constraint for pumps, but the use of inducers reduces the chore of designing for sufficient NPSP.

e. Bearing DN Speed Limit

The second speed limit encountered is roller contact bearing DN: the product of the bearing diameter and the rotational speed. DN is proportional to the tangential velocity of the bearing at the inner diameter of the inner race. Above this limit, the lubrication and cooling are insufficient. Failure will occur due to overheating, contact wear, and fatigue. Here the rotational speed must be limited to protect the integrity of material surface properties.

f. Seal Rubbing Speed Limit

The third speed limit, which is close to the DN value, is seal rubbing speed. This is the speed at which the rotating mating ring rubs against the stationary seal nose piece, causing the nose piece to wear and the seal to overheat. The turbopump rotational speed at the seal speed limit depends on the diameter. To operate at speeds exceeding this limit, special seals are required. One such is the hydrodynamic face seal (lift-off seal), which performs both the static and dynamic seal functions. When the pump is static, a spring-loaded nose presses against a mating ring to form a seal. During operation, fluid trapped in grooves forces the nose away. The nose then rides on a film of liquid and is not subject to the rubbing seal limit.

g. Centrifugal Stress Speed Limit

Speed must also be limited to protect against centrifugal stress at the roots of turbine blades. For most propellants, speeds are not high enough to make root stress a problem. Liquid hydrogen turbopumps generally do exceed the limit; thus they usually are designed at a speed below the maximum payload value. Cooling the blades with pumped propellant allows operation above the stress limit. However, thermal stresses are induced by the contrast of the hot gas on the outside and the cryogenic fluid on the inside of the blades. These stresses can cause fatigue cracking to occur. Although high speed limits must be exceeded for blade cracking to occur, the failure can be catastrophic and is worthy of concern.

h. Life Speed Limit

Operating at high speeds also compromises reliability and life. Designs for long life aim for low speeds, low stress, low temperatures, low inlet angle, low tip speed, and avoiding resonant frequencies in an attempt to lessen the severity of operating conditions. Although these criteria reduce material stresses and protect against fatigue, they also add weight, and hence lower performance. The strength and fatigue life of shafts and couplings are not a problem because the power torque loads are small relative to the shaft size. The dynamic characteristics of the shaft are more of a concern; galling, fretting, whirl, and fluctuation are frequent problems. For liquid rocket engines, pumps can operate at higher speeds because the life requirements are comparatively short.

i. Propellant Speed Limits

Properties of the working fluid substantially influence the design requirements, (headrise, NPSP, volumetric flow rate, and horsepower). The density of the fluid has the greatest effect. Because of its low density, liquid hydrogen must be pumped at high speeds. The overall headrise (pressure rise/density) of a liquid hydrogen pump is 20 times that of a liquid oxygen pump operating at the same pressure rise. This causes the top speed of the hydrogen pump to be four times that of the oxygen, assuming both are single-stage pumps. The hydrogen pump will weigh 64 times as much as the oxygen pump and will require 3 times the horsepower for operation, but the mass flowrate will be 5 times less. The difference in size of the hydrogen pump can be minimized, because the pump can be rotated faster, since the bearing and seal limits are higher. Nonetheless, to achieve the desired headrise, the speeds frequently get so high that multistaging is required to protect the bearing and seal limits. Centrifugal stress on the blades will not allow hydrogen pumps to be designed at the highest payload speed. Pumps using the denser liquid oxygen as the working fluid usually require only one stage since they need not run as fast. Unlike hydrogen though, oxygen does not have good cavitation characteristics, and, to avoid cavitation and high tank pressures, a pre-inducer is typically required.

Other properties of the working fluid influence material selection and thermal conditions of the pump. In the bearings and seals, corrosion, cooling, lubrication, and viscosity dictate the choice of materials and the speed limit. In the hot gas path, corrosion must be avoided. Conditioning the pumps includes chilling the surfaces to prevent the fluid from flashing into vapor and vapor-locking the pump. If the bearings are not cooled, they should be prechilled before contacting cryogenic fluids. To minimize the chill time and the loss of chilldown propellant, the surfaces are insulated with a thin layer of low-conductive metal. Heat soakback can be avoided by minimizing the number of pump/turbine contact points and insulating them. Because pumps in liquid rocket engines operate at such severe temperatures and pressures, it is a constant challenge to find materials that wear well during operation.

1.4.4 SSME Turbopumps

The SSME's use two-staged pumping to raise the propellant pressure from the tank storage pressure to the required chamber pressure. There are four turbopumps in each SSME: the low-pressure oxidizer and fuel turbopumps, which are axial-flow inducer-type pumps, and the high-pressure oxidizer and fuel turbopumps, which are centrifugal pumps. The low-pressure pumps are located such that the propellant enters them immediately after going through the prevalves. The high-pressure pumps are flange-attached to the hot-gas manifold, and are canted out at 10 degrees.

a. Low-Pressure Oxidizer Turbopump

The low-pressure oxidizer turbopump (LPOT), shown in figure 1.4-5 is used during engine start and mainstage to maintain sufficient pressure to the high-pressure oxidizer turbopump (HPOT) to allow it to operate at high speeds without cavitation even at worst-case inlet conditions. The LPOT is an axial-flow inducer-type pump driven by liquid oxygen tapped from the HPOT pump discharge. After driving the LPOT six-stage hydraulic turbine, the liquid oxygen passes through the turbine discharge and combines with the liquid oxygen exiting the LPOT pump discharge volutes. The combined flows are then routed to the HPOT inlet. The rotor is supported by two liquid oxygen-cooled ball bearings.

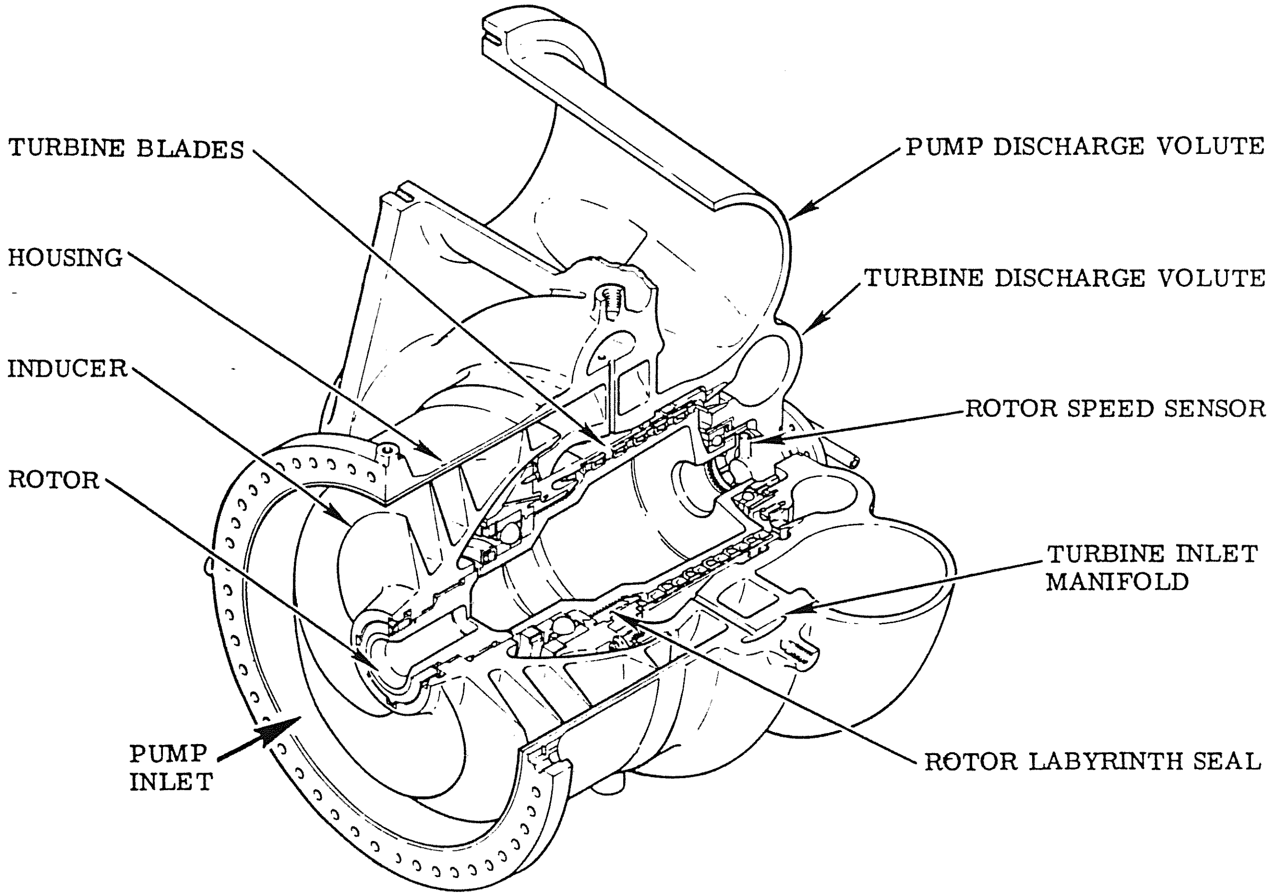


Figure 1.4-5.- Space shuttle main engine, low-pressure oxidizer turbopump.

b. Low-Pressure Fuel Turbopump

During engine start and mainstage, the low-pressure fuel turbopump (LPFT) (fig. 1.4-6) maintains sufficient pressure to the high-pressure fuel turbopump (HPFT) to permit the HPFT to operate at high speeds without an inducer and without cavitation even at worst-case inlet conditions. It is an axial-flow inducer-type pump driven by a two-stage gas turbine powered by gaseous hydrogen from the main combustion chamber coolant outlet manifold. The inducer and shaft are supported by three liquid hydrogen-cooled ball bearings. Three shaft seals are used to control leakage between the pump and turbine. Before engine start, leakage is prevented by a spring-loaded-closed, propellant-actuated-open, lift-off seal of the type described in the paragraph on the seal rubbing speed limit (sec. 1.4.3). During engine start, the seal nose is separated from its mate ring when the increasing fuel pressure overcomes the spring force. A positive separation between the seal nose and mate ring is maintained until engine shutdown when the fuel pressure decreases below the spring liftoff force.

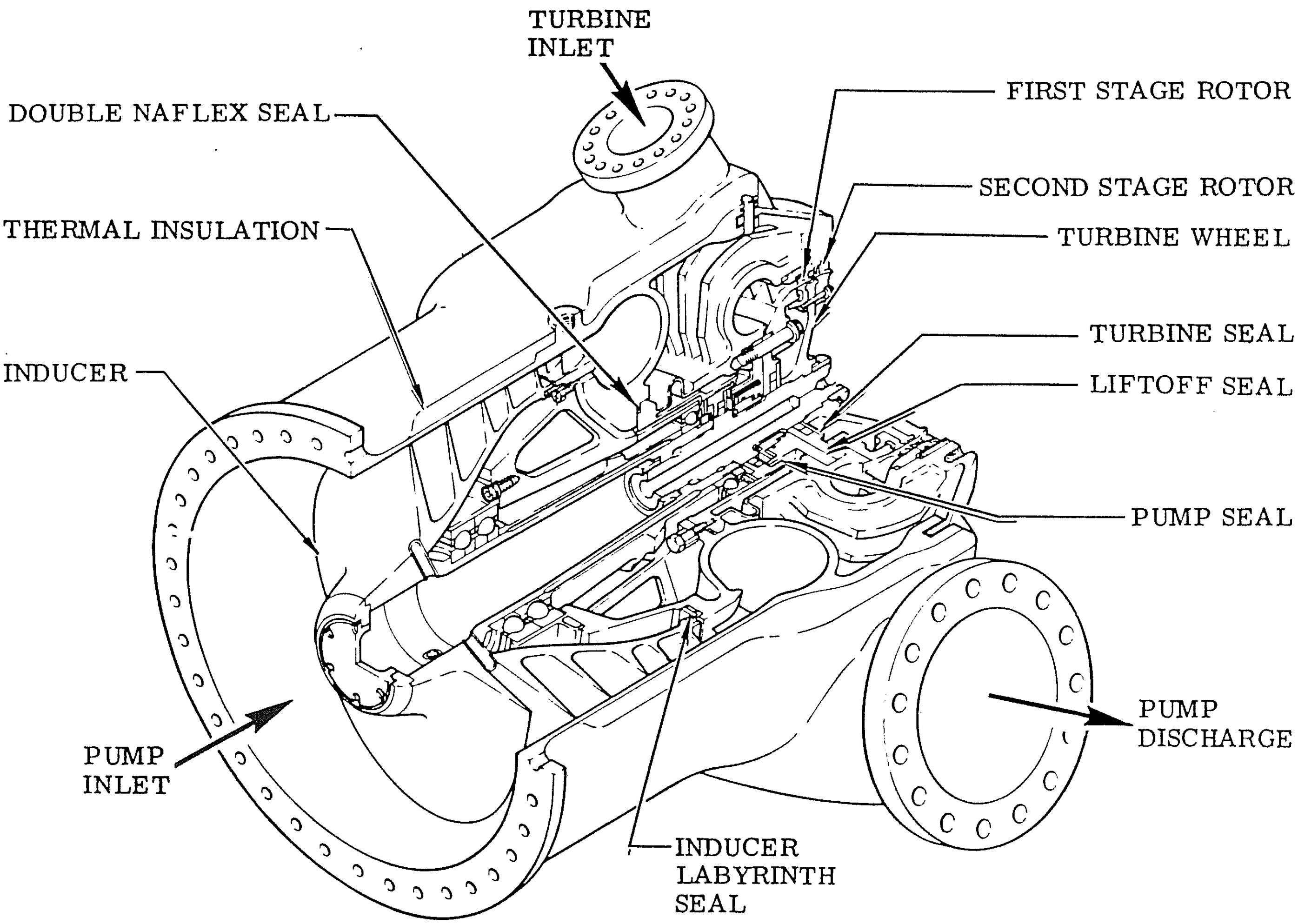


Figure 1.4-6.- Space shuttle main engine, low-pressure fuel turbopump.

c. High-Pressure Oxidizer Turbopump

The high-pressure oxidizer turbopump (HPOT) (fig. 1.4-7) receives oxidizer from the LPOT and supplies high pressure liquid oxygen to the thrust chamber injector, LPOT turbine, the heat exchanger, and the preburner boost pump. The preburner boost pump is flange-mounted to the HPOT pump end and supplies very high pressure liquid oxygen to the oxidizer pre-burner and fuel pre-burner. Hydrogen-rich steam from the oxidizer pre-burner powers the turbine which drives the HPOT. The main pump consists of two single-stage centrifugal pumps on a common shaft. The HPOT has a single inlet with a 50-50 flow split into a double-entry, common outlet impeller. The four HPOT shaft bearings are cooled by liquid oxygen from the preburner pump. All components of the turbine end are cooled by gaseous hydrogen flowing over or through them. Mixing of liquid oxygen from the pump end and hydrogen-rich steam from the turbine end (a very explosive combination) is prevented by a dynamic shaft seal package that is between the main pump and the turbine (fig. 1.4-8). The seal package consists of a labyrinth-type primary oxidizer seal, a helium purge flow controlled-gap intermediate seal, and two controlled-gap turbine hot-gas seals. The helium purge in the intermediate seal area mixes with and flushes overboard any leakage of liquid oxygen past the labyrinth seal and any leakage of hydrogen-rich steam past the hot-gas seals.

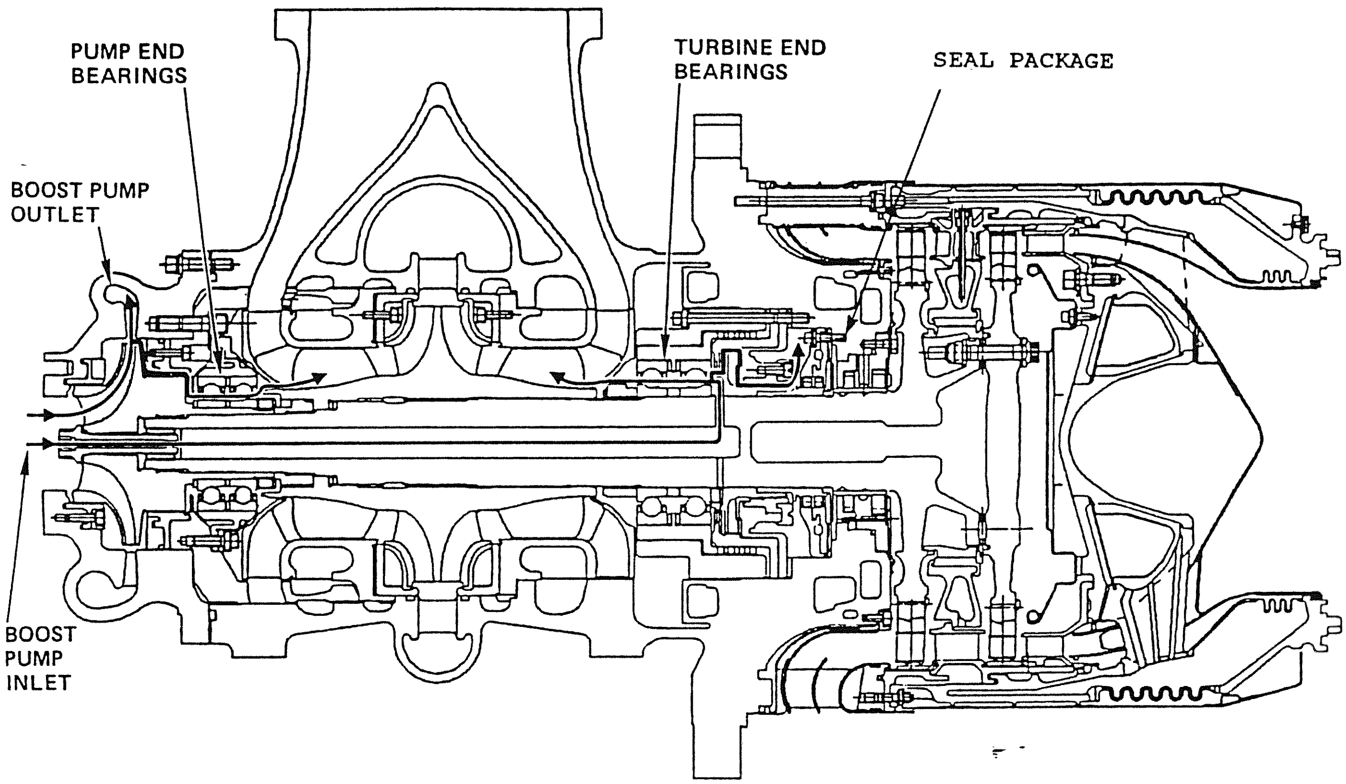


Figure 1.4-7.- Space shuttle main engine, high-pressure oxidizer turbopump.

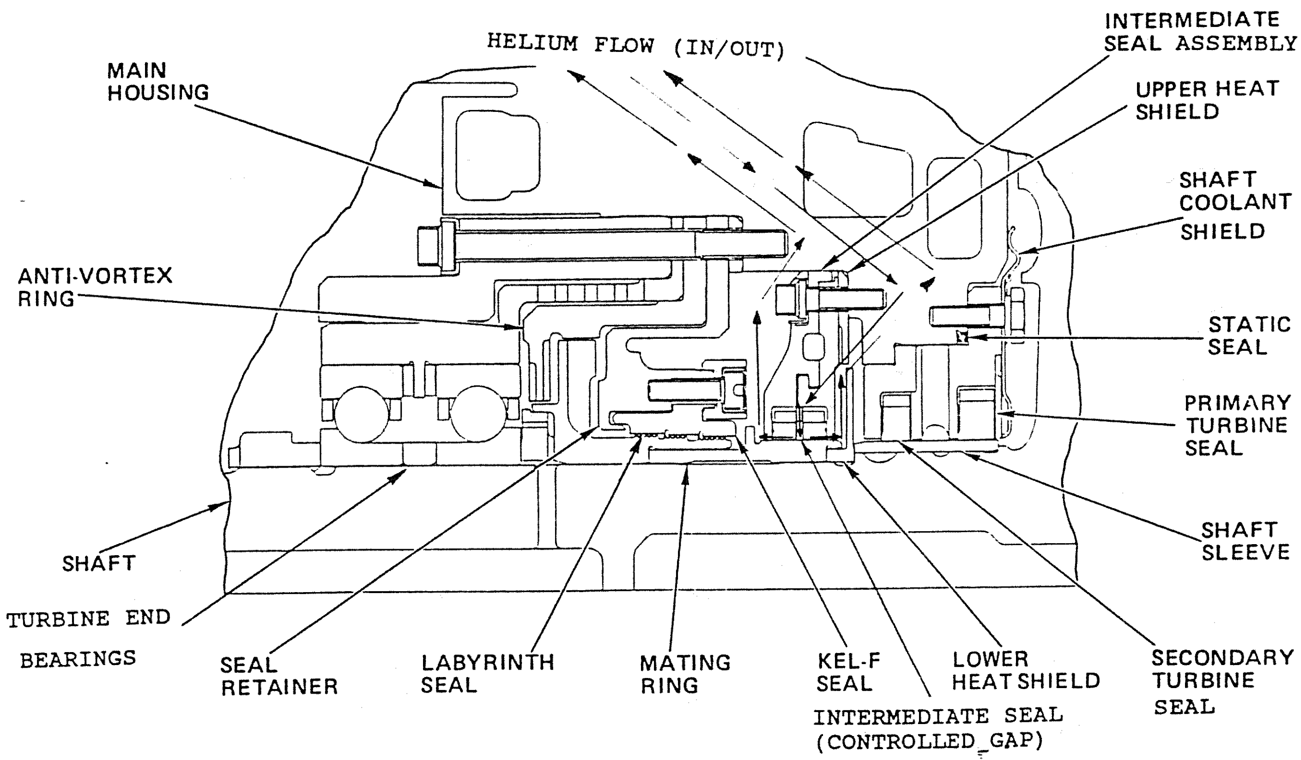


Figure 1.4-8.- Space shuttle main engine, high-pressure oxidizer turbopump shaft seal package.

d. High-Pressure Fuel Turbopump

The HPFT (fig. 1.4-9) receives fuel from the LPFT and supplies it at increased pressure through the main fuel valve to the thrust chamber assembly coolant circuits. The HPFT is a three-stage centrifugal pump that is directly driven by a two-stage turbine powered by hydrogen-rich steam from the fuel preburner. Fuel flows in series through the three impellers from the pump inlet to the pump discharge with flow being redirected between the impellers by interstage diffusers. The seals are similar to those on the HPOT with the helium purge seal being replaced by a liftoff seal on the HPFT. The liftoff seal on the HPFT maintains positive flow of liquid hydrogen from the pump end towards the turbine end bearings. Any leakage of hydrogen-rich steam past the labyrinth seal is entrained in this flow. Hydrogen-rich steam can safely mix with liquid hydrogen since this is not an explosive mixture (unlike liquid oxygen mixing with hydrogen-rich steam in the HPOT, which is highly explosive). During engine start, the liftoff seal nose is separated from its mate ring when increasing fuel pressure overcomes the spring force. A positive separation between the seal nose and mate ring is maintained until engine shutdown, when the actuating pressure decreases below the spring force. Propellant flow through the seal and mate ring is used to cool the turbine-end bearings and the turbine components (fig. 1.4-10). Coolant flow across the pump-end bearings is provided by the first-stage impeller back-plate wear-ring flow.

During all phases of SSME operation where the HPFT is rotating, the HPFT balance cavity maintains the proper axial position of the HPFT shaft. During start and shutdown transients, the balance cavity is assisted by the thrust bearing. The balance cavity on the HPFT is both the front and back faces of the third-stage impeller. The pressure in these two separate cavities is regulated by orifices fed at the inlet to and the exit from the third-stage impeller. As the shaft moves toward the pump end, the orifices at the exit of the pump open on the front face and close on the rear face. Meanwhile, the orifices at the entrance to the pump close at the front face and open at the back face (the back face "entrance" is really the exit from the balance cavity to the turbine coolant loop). The end result is that the pressure acting on the front face increases, while the pressure on the rear face decreases. The force imbalance pushes the shaft back toward the turbine end and, in the process, opens and closes the orifices the reverse of previously. In this manner, the balance cavity also controls shaft movements toward the turbine end. The HPOT has a similar balance cavity arrangement, with the "front" faces on both the left and right side impellers being the opposing balance cavities.

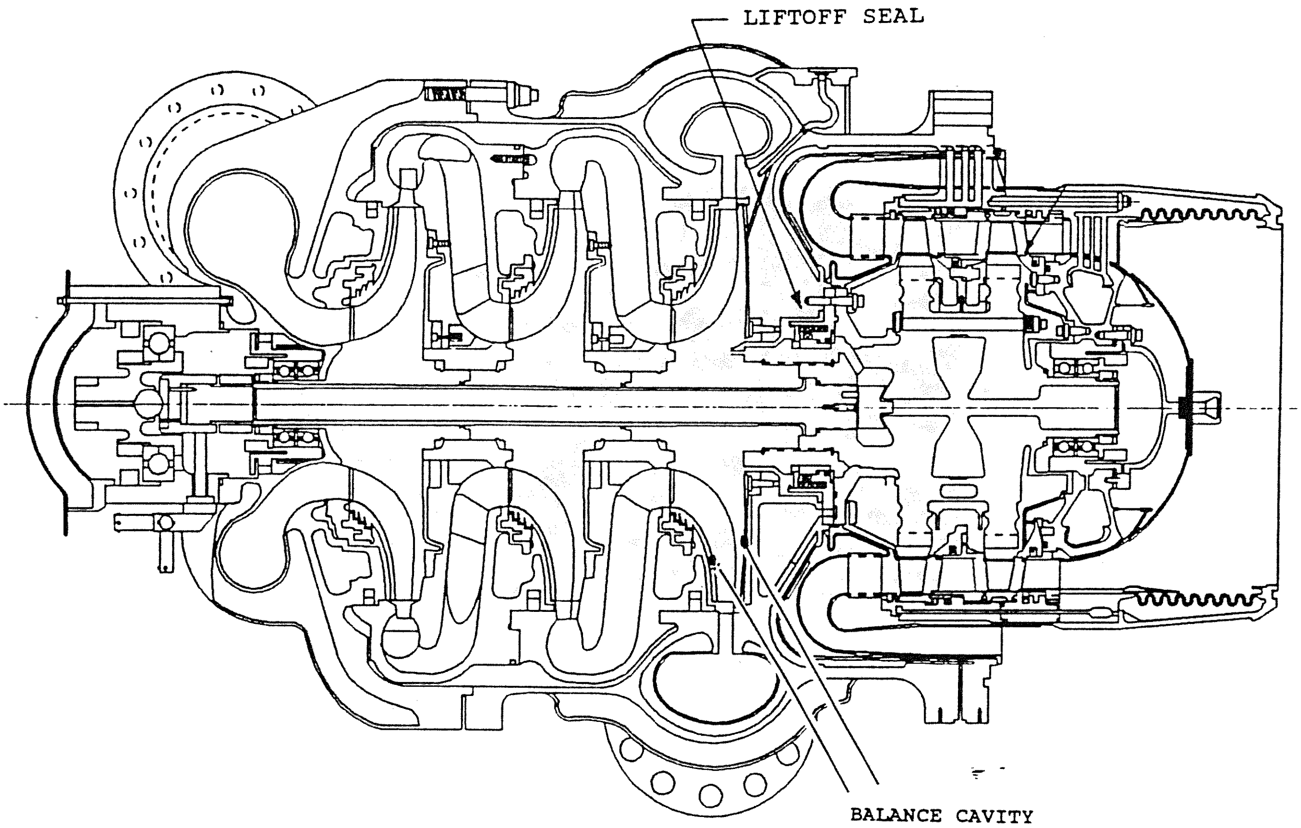


Figure 1.4-9.- Space shuttle main engine, high-pressure fuel turbopump.

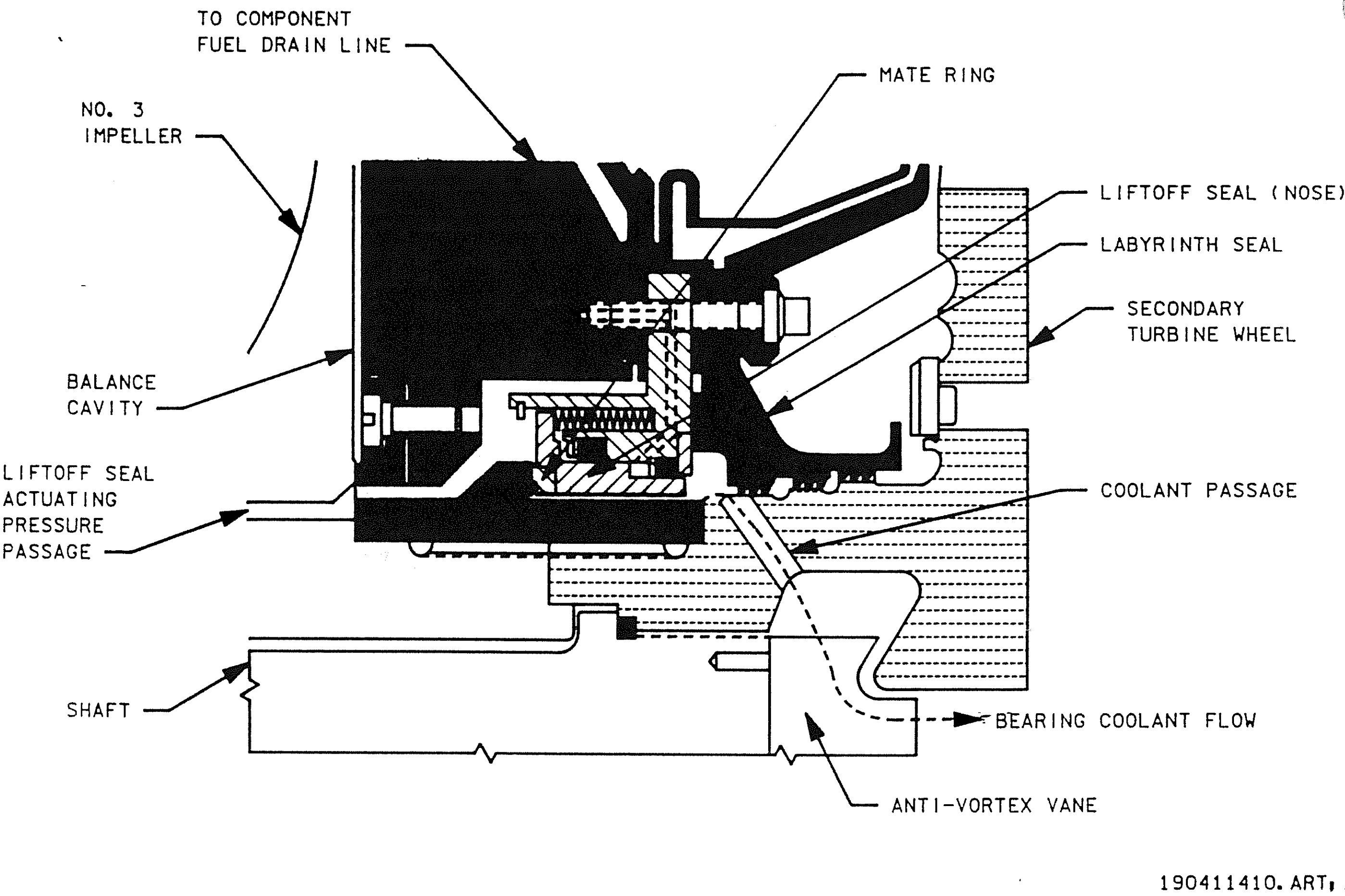


Figure 1.4-10.- Space shuttle main engine, high-pressure fuel turbopump shaft seal package.

1.4.5 References

Bathie, William W. Fundamentals of Gas Turbines. John Wiley and Sons, New York. 1984.

Wilson, David G. The Design of High-Efficiency Turbomachinery and Gas Turbines. The MIT Press, Cambridge, Massachusetts. 1984.

Hill, Philip G. and Peterson, Carl R. Mechanics and Thermodynamics of Propulsion. Addison-Wesley Publishing Company, Reading, Massachusetts. 1965.

NASA SPACE VEHICLE DESIGN CRITERIA (CHEMICAL PROPULSION)

NASA SP-8107: Turbopump Systems for Liquid Rocket Engines

NASA SP-8109: Liquid Rocket Engine Centrifugal Flow Turbopumps

NASA SP-8125: Liquid Rocket Engine Axial Flow Turbopumps

NASA SP-8110: Liquid Rocket Engine Turbines

NASA SP-8052: Turbopump Inducers

NASA SP-8101: Turbopump Shafts and Couplings

SSME Operation and Maintenance Documentation
E41000-8
RSS-8559-2-1
Volume I
General Information

SSME Orientation
Part A - Engine
Rocketdyne Course
No. ME-110 (A)RIR

1.5 SSME MANUAL THROTTLING/MAXIMUM THROTTLES

1.5.1 Manual Throttling

Introduction - The SSME throttle commands are nominally generated by ascent guidance and sent to the SSME SOP software sequence. The SSME SOP then sends the commanded SSME throttle setting to all three engine controllers. However, for certain cases the throttling capability must be taken over manually. When manual throttling is engaged, the pilot can throttle the SSME's by matching the current automatic throttle setting and then moving the right hand speed brake/throttle controller (SBTC) handle. The SBTC SOP generates the throttle settings in percent. When manual throttling is engaged the following automatic functions are lost:

a. Throttle up to mission power level after liftoff. (Recall that liftoff is performed at 100 percent power level.)

b. Throttling for maximum dynamic pressure when three SSMEs are operating

c. Possible throttling to the abort power level (i.e., 109 percent) after one engine out

d. Guidance switching from AOA to nominal MECO targets in second stage

e. Throttling to limit vehicle acceleration below 3g

f. Throttling at fine count to minimum power level, which occurs 6.871 seconds prior to MECO

g. Guidance-commanded MECO

h. Fuel dissipation throttling during RTLS flyback

i. Maximum throttle (i.e. 109 percent) cannot be engaged

When possible, throttle control needs to be returned to auto control, because all of the above automatic functions are regained except for function h. The throttle control will not be returned to automatic control if manual guidance is selected. This special case is discussed in section 1.5.1.2.

The low-level cutoff capability is still available when manual throttling is engaged because guidance is not involved; however, the SSME's may not be at the proper throttle setting for cutoff because function f. above is not active while manual throttling is engaged.

1.5.1.1 Procedural Description for Manual Throttling

The following steps describe how manual throttling is engaged and disengaged:

a. The manual throttling capability is supported by PASS but not by BFS.

b. The manual throttling capability is restricted to the right hand speed brake/throttle controller (RH SBTC) only.

c. The pilot depresses and holds the takeover button located on the right side of the SBTC handle on panel C3. Depressing the SBTC takeover button extinguishes the AUTO lamps of the SBTC pushbutton indicators on panels F2 and F4 and freezes the throttle command at its last value before takeover.

d. The pilot moves the SBTC handle forward or aft, with the takeover button still depressed, until the takeover manual command matches the last auto throttle command. (If the throttle command is not matched before the takeover button is released, the throttle control will be returned to auto throttling and the AUTO SBTC lamp will be illuminated.)

e. When the manual throttle setting matches the auto throttle setting, the MAN lamp of the RH SBTC pbi on panel F4 will be illuminated.

f. With manual throttling engaged, the SBTC takeover button can be released.

g. Subsequent movements of the SBTC handle will generate corresponding throttle commands to all three SSME's. The speed brake/throttle controller compensated position is converted into a throttle setting in percent rated power level (RPL) and is limited to a value within the minimum and maximum throttle command range. The minimum and maximum throttle commands are I-loads. Nominally the minimum throttle setting (K_MIN) is 67 percent. The maximum throttle (K_MAX) setting, which corresponds to the SBTC handle being in the full forward position, is set at the nominal flight throttle command which is normally 100 or 104 percent. If the emergency throttle command, which is 109 percent, is set by crew item entry on the Override display, K_MAX is then set equal to 109 percent and the SBTC is rescaled. The full forward position will then be 109 percent.

h. Depressing the takeover button during manual throttling has no affect.

i. To return to auto throttling from manual control, either the LH or RH SBTC pbi (on panels F2 and F4, respectively) is depressed, which illuminates the AUTO lamp. Throttle matching is not required before returning to auto throttle control. As a result, the throttle setting will remain at the power level set during manual throttling until one of the automatic functions described above in items a. through i. paragraph 1.5.1 is commanded.

j. If manual throttling is engaged and maintained until MECO, the SSME's must be manually throttled for maximum dynamic pressure, for 3g limiting, and to the fine count velocity. The SSME's must be manually shut down at MECO. Recall low level cutoff is independent of guidance, therefore, the low level cutoff capability is still available. However, the power level may not correspond to the low level timer that would be used because guidance is what commands the engines to the cutoff velocity.

1.5.1.2 Cases Requiring Manual Throttling

The flight rules in the Flight operations, Guidance and Booster sections identify cases that require manual throttling. Some of the cases include the following:

a. Anytime manual guidance is selected (i.e., control stick steering). The reason for this action is that since guidance cannot be trusted to steer the vehicle, guidance cannot be trusted to command the engines. Therefore, the crew will have to throttle the engines manually for all functions such as 3g throttling and fine count, and manually shut down the engines at MECO.

b. Failure of all three SSME's to throttle for control of vehicle dynamic pressure and acceleration limits.

c. Initiation of a three-SSME TAL abort or a two-SSME TAL abort with inertial velocity greater than 13.7K fps. This is done to maximize the length of time to do the OMS dump.

d. Low LH_2 NPSP.

Rationale for these and other manual throttle rules are covered in the flight rule rationale document.

1.5.1.2 SBTC Takeover Switch and Pushbutton Redundancy

The takeover switch has three discretes. Thee is no fault detection, identification, or reconfiguration (FDIR) performed on the SBTC takeover switch during Ops 1 or 6.

The three commands from the SBTC transducers are handled by the selection filter in the same way as the SRB chamber pressure transducers are handled. If all three commands are valid, the midvalue select is used; if two commands are valid, the average is used; if one command is valid, that value is used; and if no valid command is available, the previous valid value will be used.

To return to auto throttling, either the LH or RH SBTC pbi can be depressed. Each of these pbi's is a three contact switch.

1.5.2 Maximum Throttles

To set the maximum throttle command in the PASS or BFS to 109 percent, the crew must enter item 4 on the override display. Guidance will set the throttle command equal to the maximum throttle command (i.e. engage maximum throttles for all three engines) if manual throttling is not engaged and if the following conditions are met:

a. In MM102, at least one SSME must have failed.

b. In MM103, the 3g limiting throttling must not be in progress.

c. In MM601, during the fuel dissipation phase (i.e., before powered pitcharound), at least one SSME must have failed.

During the flyback phase (i.e., after powered pitcharound) in MM601, the engines will be commanded between 67 and 109 percent as required to meet the guidance powered pitchdown targets, regardless of the number of engines operating.

When maximum throttles are engaged, the SBTC is rescaled with the full forward position at the maximum power level setting.

1.5.3 References

FSSR - STS 83-0015A - Controllers

FSSR - STS 83-0002A - Ascent/RTLS Guidance

Flight Rule Rationale

1.6 SSME VALVE SCHEMATICS

1.6.1 General

The SSME's are liquid propellant rocket engines; therefore, valves are required to control the flow of propellants. The five primary valves are as follows:

- Oxidizer preburner oxidizer valve (OPOV)
- Fuel preburner oxidizer valve (FPOV)
- Chamber coolant valve (CCV)
- Main fuel valve (MFV)
- Main oxidizer valve (MOV)

The pogo suppression system has the following valves:

- Pogo recirculation isolation valve (RIV)
- Pogo helium precharge valve (HPV)
- Pogo gaseous oxygen control valve (GCV)

The remaining SSME valves described in this systems brief include:

- Propellant bleed valves
- Antiflood valve (AFV)
- Helium purge check valve
- Normally open solenoid valve
- Normally closed solenoid valve
- Pressure actuated valves
- Propellant valve hydraulic actuators

1.6.2 Functional Description

1.6.2.1 Oxidizer Preburner Oxidizer Valve

The OPOV is shown in figure 1.6-1. The OPOV is a retracting seal ball valve with a rectangular flow passage. A flow "slot" as opposed to a flow "hole" makes valve opening versus ball rotation a linear function. The OPOV, which is flange mounted between the oxidizer supply line to the oxidizer preburner (OPB) and the OPB oxidizer inlet, controls oxidizer flow to the OPB and the OPB augmented spark igniter (ASI). At engine ignition, the OPOV is ramped open to start flow and continues ramping through thrust buildup. During ascent, it is modulated to control engine thrust between minimum power level (MPL) and full power level (FPL).

The OPOV assembly consists mainly of three moving components: a shaft/ball assembly, a cam follower, and a bellows/ball seal. The ball, cam, and shaft are connected to form a rigid throttling spool. The ball inlet seal is a machined plastic, bellows-loaded, closed seal. Redundant shaft seals, with

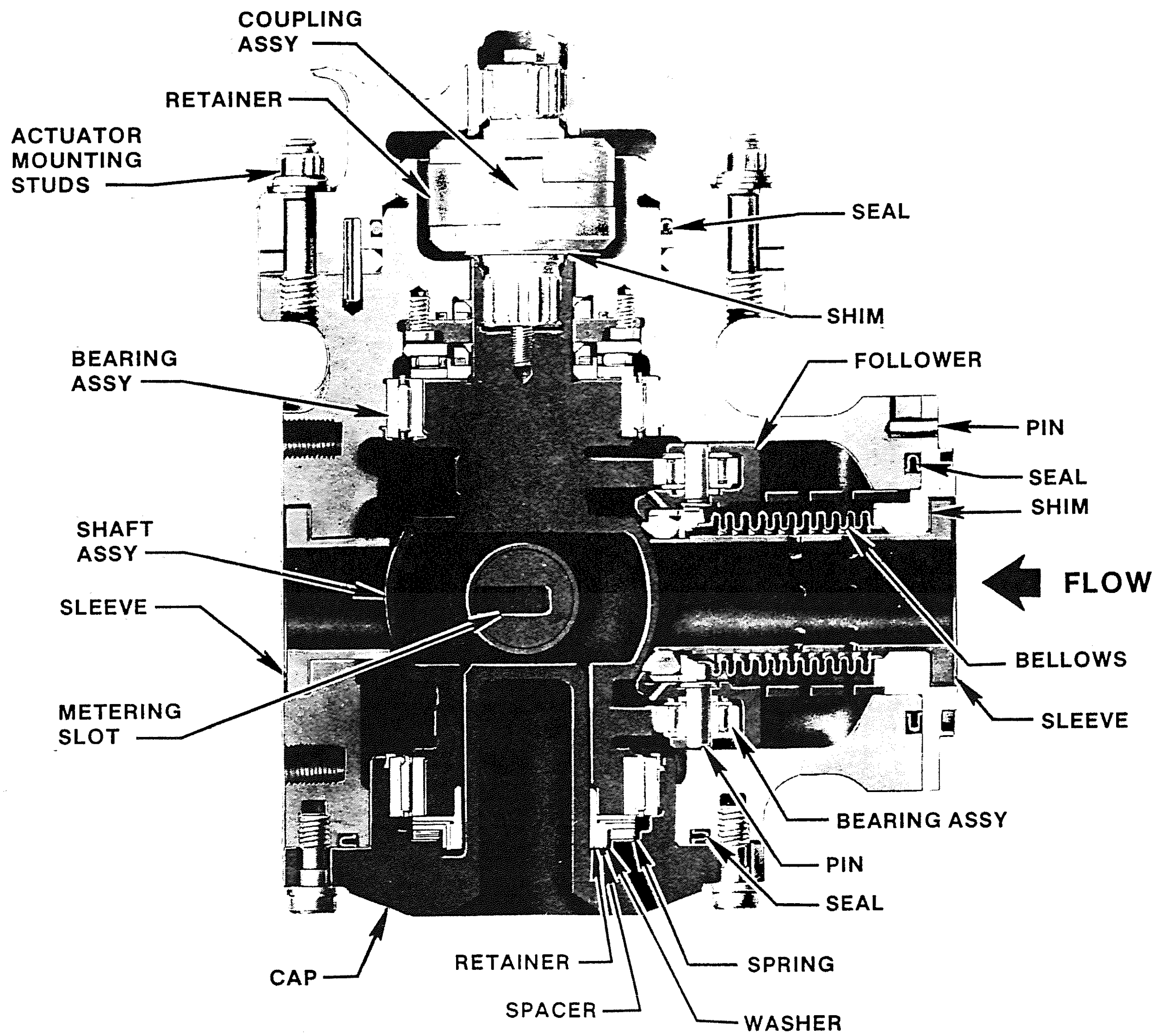


Figure 1.6-1.- OPOV.

an overboard drain between them, prevent leakage along the shaft (actuator end) during engine operation. Inlet and outlet throttling sleeves align the flow to minimize turbulence and the resultant pressure loss. Ball seal wear is minimized by cams and a cam follower assembly that displace the seal from the ball during the initial degrees of ball travel. During initial opening of the valve, the shaft rotates 15° and actuates the cam mechanism to retract the ball seal. Retracting the ball seal from the ball allows oxidizer flow around the ball and through the clearance space of the downstream seal, providing oxidizer flow to the OPB ASI. During the last 15° of shaft motion, the cam follower rides down the cam and allows the ball seal to be held against the ball by the spring load of the bellows/cam follower. The OPOV dimensions are shown below.

Flow passage dimensions	1.100 x 0.374 in.
Ball diameter	2.968 in.
Shaft diameter	1.100 in.
Valve length	6.812 in.

The OPOV is actuated by a hydraulic actuator that mounts to the valve housing and transmits rotary force to the shaft coupling.

1.6.2.2 Fuel Preburner Oxidizer Valve

The FPOV (fig. 1.6-2) is a retracting seal ball valve with a circular flow passage. The flow "hole" makes variation of fuel flow a nonlinear function over 1 percent increments in valve position. Flange mounted between the supply line to the fuel preburner (FPB) and the FPB inlet, the FPOV controls oxidizer flow to the FPB and the FPB ASI. At engine start, the FPOV is ramped open to provide fuel for combustion and continues ramping to control the mixture ratio through thrust buildup. In mainstage, the FPOV is modulated to control the mixture ratio to 6.011 LOX/FUEL.

The FPOV assembly consists mainly of three moving components: a shaft/ball, a cam follower, and a bellows/ball seal. The ball inlet seal is a machined plastic, bellows-loaded, closed seal. Redundant shaft seals, with an overboard drain between them, prevent leakage along the shaft (actuator end) during engine operation. Inlet and outlet throttling sleeves align the flow to minimize turbulence and the resultant pressure loss. Ball seal wear is minimized by cams and a cam follower assembly that displace the seal from the ball during the initial degree of ball travel. During initial opening of the valve, the shaft rotates 15° and actuates the cam mechanism to retract the ball seal. Retracting the ball seal from the ball allows oxidizer flow around the ball and through the clearance space of the downstream seal, providing oxidizer flow to the FPB ASI. The FPOV dimensions are shown below.

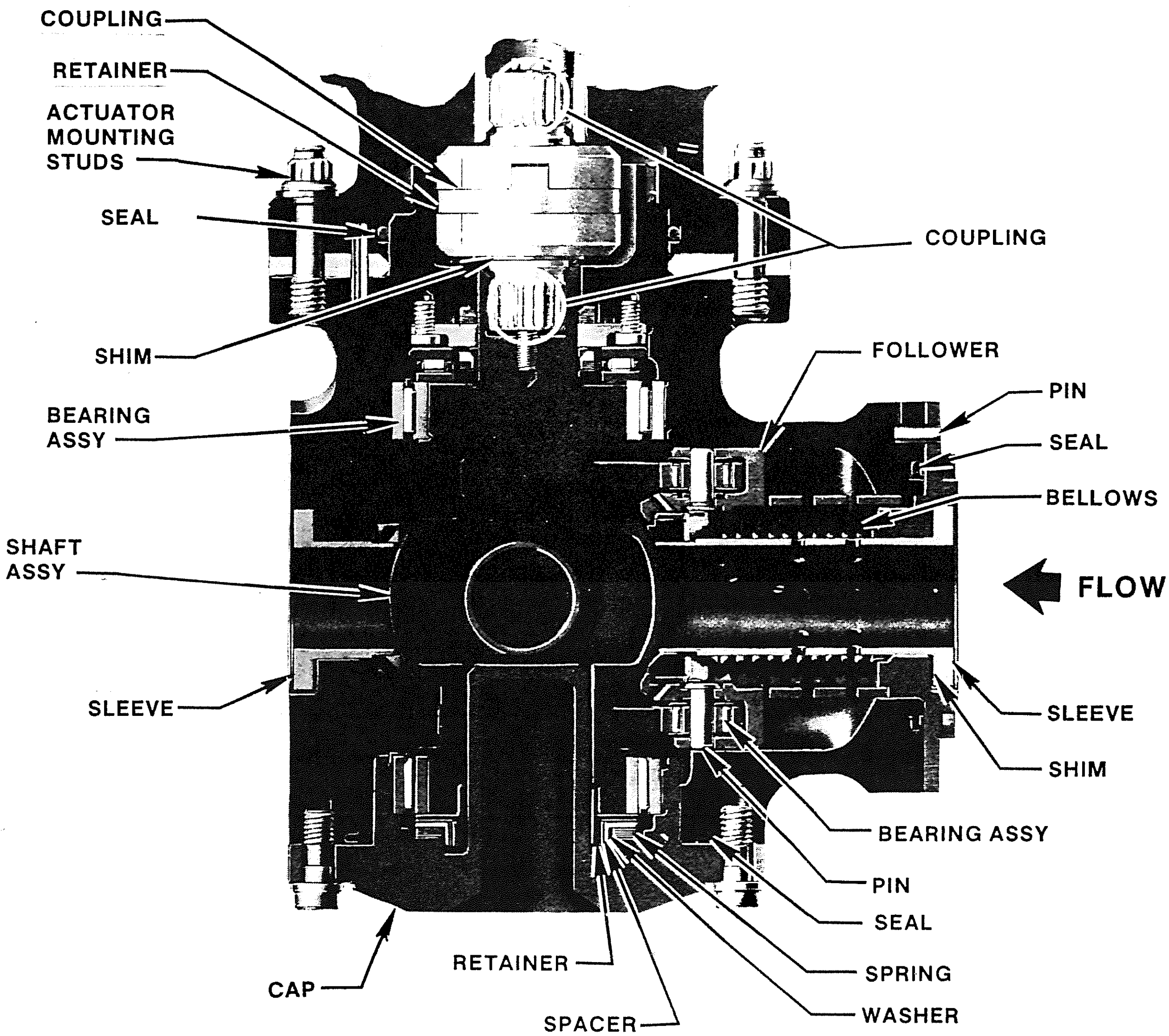


Figure 1.6-2.- FPOV.

Flow passage diameter	1.100 in.
Ball diameter	2.968 in.
Shaft diameter	1.100 in.
Valve length	6.812 in.

The FPOV is actuated by a hydraulic actuator that mounts to the valve housing and transmits rotary force to the shaft coupling.

1.6.2.3 Chamber Coolant Valve

The chamber coolant valve is shown in figure 1.6-3. It is a hydraulically actuated, gate-type valve that regulates fuel flow through the main combustion chamber, nozzle, and turbopump coolant circuits. The valve is installed as an integral component of the nozzle forward manifold assembly, the CCV duct. The CCV does not have a gate seal because it is located downstream of the main fuel valve (MFV) and is not required to be a positive shutoff valve. Redundant shaft seals, with an overboard drain cavity between them, prevent leakage along the shaft during engine operation. The CCV dimensions are shown below.

Flow passage diameter	1.600 in.
Gate diameter	2.5 in.
Shaft diameter	1.100 in.
Valve length	7.729 in.

The CCV position is varied linearly with engine chamber pressure such that it is 50 percent open at MPL and 100 percent open at rated power level (RPL).

1.6.2.4 Main Fuel Valve

The MFV is shown in figure 1.6-4. The MFV is a ball valve with a 2.5-inch propellant flow passage, a hydraulic actuator in response to commands from the engine controller. The MFV is flange-mounted between the high pressure fuel duct and the coolant inlet distribution manifold of the thrust chamber nozzle. The valve controls the flow of fuel from the HPFT to the coolant inlet distribution manifold of the thrust chamber assembly. The MFV is actuated to a full-open position at start, scheduled full-open during mainstage operation, and actuated to full-close position at engine shutdown.

The MFV basically has three moving components: a shaft/ball assembly, a cam follower, and a bellow/ball seal. The ball, cam, and shaft are connected to form a rigid throttling spool. Redundant shaft seals, with an overboard drain cavity between them, prevent leakage along the shaft (actuator end)

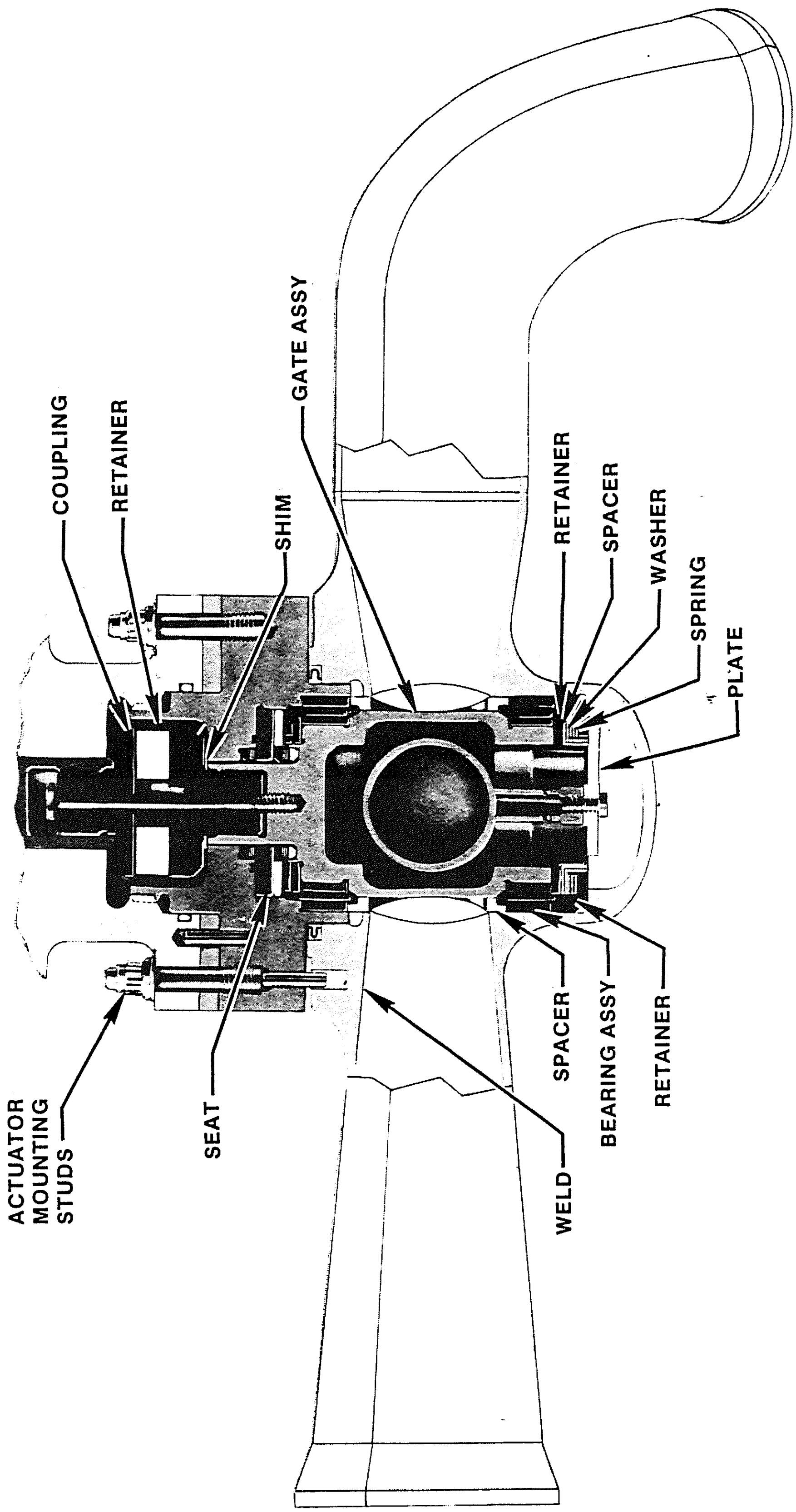


Figure 1.6-3.-CCV.

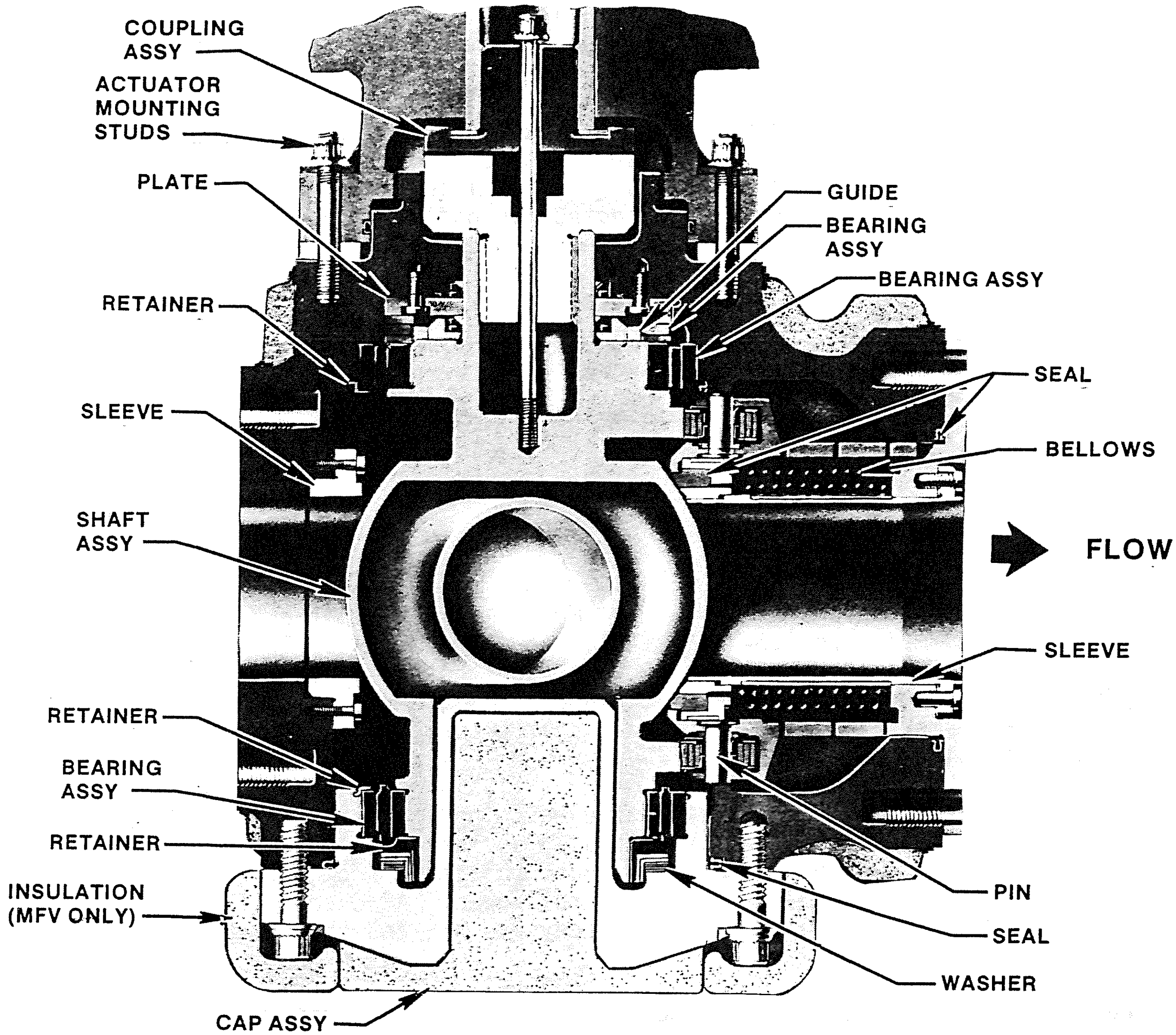


Figure 1.6-4.- MFV.

during engine operation. Inlet and outlet throttling sleeves align the flow to minimize turbulence and the resultant pressure loss. During initial opening of the valve, the shaft rotates approximately 10° and actuates the cam mechanism to lift the ball seal. After this initial rotation, the ball seal is fully retracted. During the final portion of valve closing the cam follower rides down ramps on the cams and allow the ball seal to be held against the ball by the spring load of the bellows.

The MFV hydraulic actuator mounts to the valve housing and transmits rotary force to the shaft coupling. Dual shaft seals in both the valve and actuator, with vents between the seals, control leakage into the coupling cavity. The coupling cavity is protected against overpressurization by a burst diaphragm. The MFV dimension are shown below.

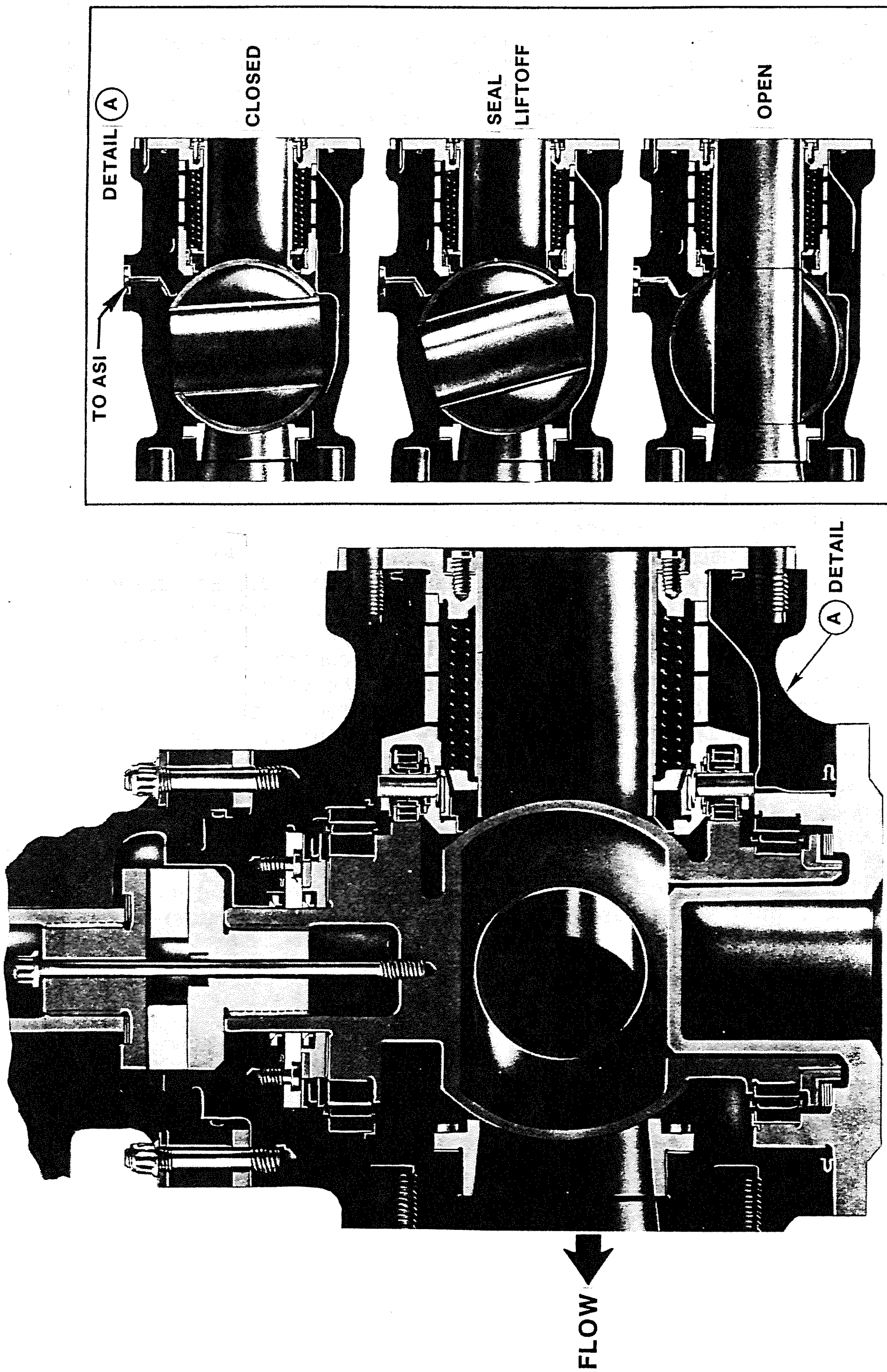
Flow passage	2.500 in.
Ball diameter	5.125 in.
Shaft diameter	1.875 in.
Valve length	10.25 in.

1.6.2.5 Main Oxidizer Valve

The MOV is shown in figure 1.6-5. The MOV is similar to the MFV and the parts labeled in figure 1.6-4 also apply to figure 1.6-5. The MOV is a ball valve with a 2.5-inch propellant flow passage that is actuated by a hydraulic actuator in response to commands from the engine controller. The MOV is flange-mounted between the main chamber oxidizer dome and the high pressure oxidizer duct.

The valve controls oxidizer flow to the main injector and main injector (ASI). At engine start, the MOV initially is ramped open to start ignition flow, then scheduled open as engine thrust builds up from MPL to RPL. During mainstage, the MOV is scheduled full-open. At cutoff, the MOV is scheduled closed until engine thrust decays to MPL and then ramped to full-close.

The MOV basically has three moving components: a shaft/ball assembly, a cam follower, and a bellow/ball seal. The ball, cam, and shaft are connected to form a rigid throttling spool. During initial opening of the valve, the shaft rotates approximately 10° and actuates the cam mechanism to lift the ball seal. Retracting the ball seal from the ball allows oxidizer flow through the clearance space around the ball, providing oxidizer flow to the main injector ASI. After this initial shaft rotation, no further motion of the ball seal occurs. During the final portion of valve closing, the cam follower rides down ramps on the cams and allows the ball seal to be held against the ball by the spring load of the bellow.



(A) The MOV is very similar to the MFV (Figure 1.6-4). Parts labled in Figure 1.6-4 also apply here.

Figure 1.6-5.- MOV.

The MOV hydraulic actuator mounts to the valve housing and transmits rotary force to the shaft coupling. Dual shaft seals in both the valve and the actuator, with vents between the seals, prevent leakage into the coupling cavity. The coupling cavity is protected against overpressurization by a burst diaphragm. The MOV dimensions are shown below.

Flow diameter	2.5 in.
Ball diameter	5.125 in.
Shaft diameter	1.875 in.
Valve length	10.25 in.

1.6.2.6 Pogo Recirculation Isolation Valve

The RIV is shown in figure 1.6-6. The RIV is located in the Pogo accumulator overflow discharge to the oxidizer bleed line. The RIV is closed during engine thermal conditioning to prevent short-circuiting of the oxidizer bleed circulating flow through the accumulator. The RIV opens upon command, allowing discharge of excess liquid oxygen (LOX) and gaseous oxygen (GOX) from the accumulator to control the accumulator gas/liquid level. The normally open RIV is pneumatically actuated closed using a bellows actuator and opened by the bellows spring force when the control pressure is vented. The RIV incorporates a GOX override that opens the valve even if the control pressure is not vented. GOX pressurant is admitted to the underside of the actuator and forces the valve open against the pneumatic control pressure. A drain between the actuator shaft seals precludes leakage of the high-pressure GOX into the oxidizer bleed line.

The RIV incorporates an LVDT position indicator that converts valve position to an electrical signal. The controller monitors the electrical signal to verify valve position during the propellant conditioning and engine start phases. The signal serves as a maintenance recording parameter during the remaining phases.

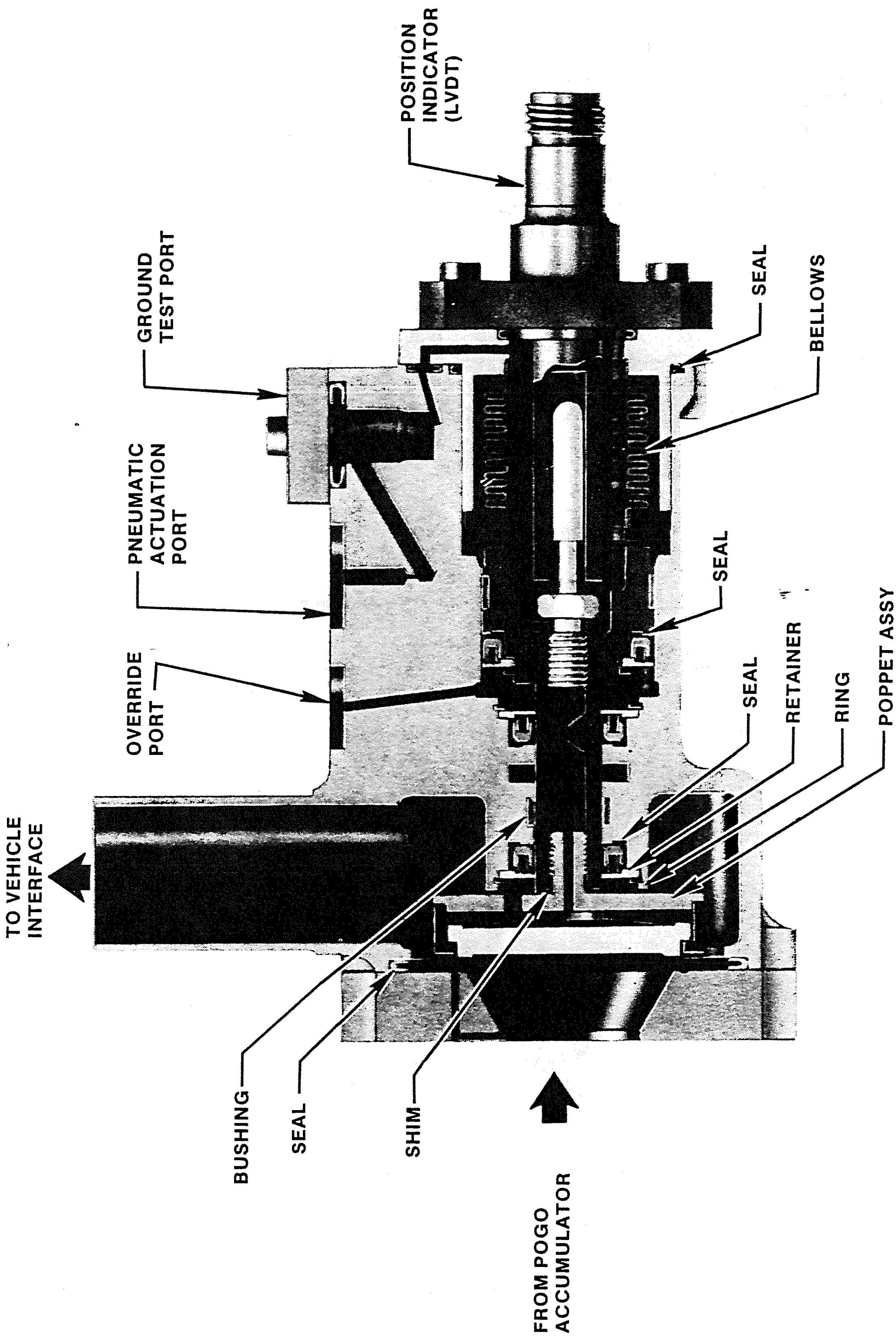


Figure 1.6-6.- RIV

1.6.2.7 Pogo Helium Precharge Valve

The schematic for the HPV is shown in figure 1.6-7. The HPV assembly provides helium pressurant to the Pogo accumulator during engine start until GOX is available from the engine heat exchanger. The HPV is also used to provide helium to the accumulator as a post charge at engine shutdown.

The HPV is a spring-loaded closed, pneumatically actuated open poppet valve. It has an integral check valve at its outlet port. The helium supply is filtered by a 15-micron absolute filter at the HPV inlet.

A dual-coil, normally closed solenoid valve is mounted on the HPV and provides pilot control of the normally closed HPV and the normally open GCV.

During the engine start preparation phase, the HPV body is warmed by the main injector oxidizer dome nitrogen purge gas that flows through the HPV body before being routed to the main injector.

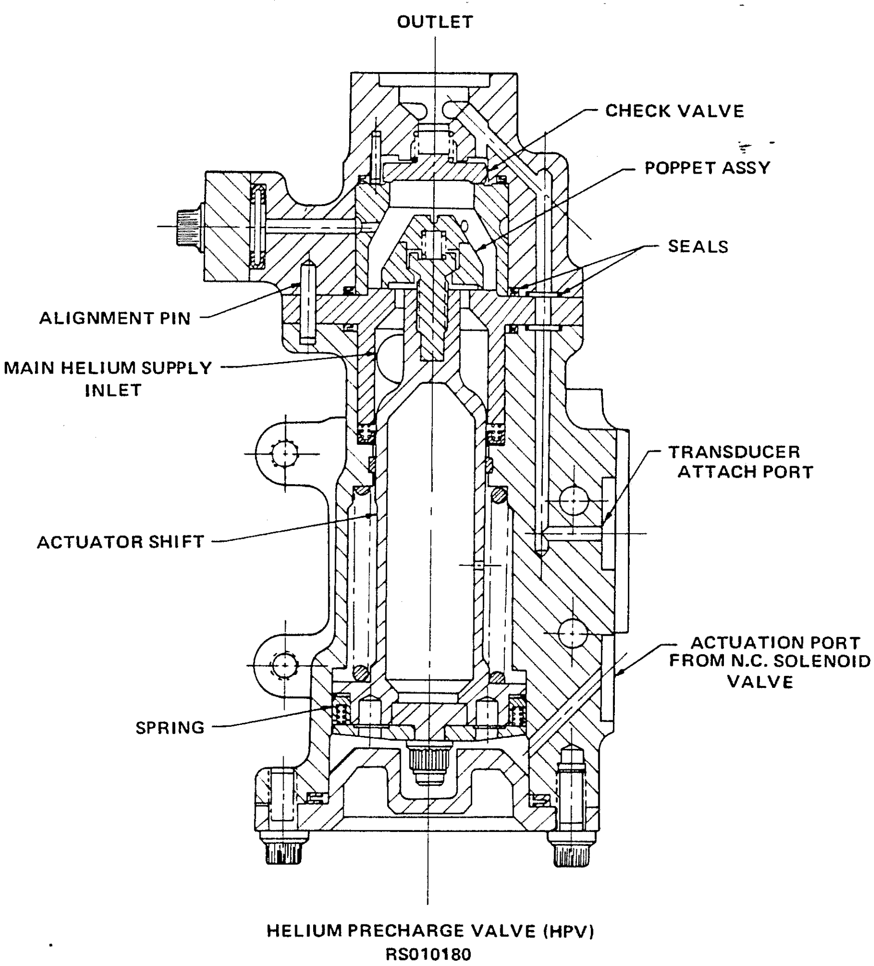


Figure 1.6-7.- Pogo helium precharge valve.

1.6.2.8 Pogo Gaseous Oxygen Control Valve

The GCV is shown in figure 1.6-8. Pogo suppression is accomplished by a gas-filled accumulator in the LOX feed at the HPOT inlet. GOX is used as the compliant medium subsequent to an initial helium precharge. During engine start this helium precharge ensures rapid charging of the accumulator to provide Pogo protection during lift-off and the early part of boost. GOX generated in the heat exchanger is used to maintain the accumulator charge during the remainder of boost.

The GOX control valve assembly prevents reverse LOX flow from the accumulator from entering the heat exchanger during engine thermal conditioning and blocks GOX flow during engine start to enable the helium precharge.

The GOX control valve assembly combines a pressure-actuated GOX supply valve and a check valve in the same housing.

The GOX supply valve is a poppet-type valve, pressure-actuated closed by application of pneumatic control pressure on a piston, and bellows spring-loaded to its normally open position when the control pressure is vented. A bleed orifice from the closing to the opening side of the actuator piston provides for automatic opening of the valve 3 to 10 seconds after control pressure is applied. The poppet pivots so it can align itself with the seat. The RIV is maintained in the open position during engine operation by connecting GOX pressure from the GCV to the override port of the RIV.

1.6.2.9 Propellant Bleed Valves

The oxidizer bleed valve (OBV) and the fuel bleed valve (FBV) are the same type of valve, which is illustrated in figure 1.6-9.

The OBV and FBV are spring- and bellows-loaded closed, pneumatically-actuated open, metal-to-metal seat poppet valves. The valves are open by pneumatic pressure from the pneumatic control assembly (PCA) during engine start preparation. This provides a recirculation flow for propellants through the engine to ensure that propellants in the engine are at the required temperatures for engine start. At engine start, the valves are closed by venting the actuation pressure. The valves are fail-safe in that pressure acting on the unbalanced area poppet, combined with spring and bellows forces, can overcome the actuation pressure. The valves have LVDT's for full-open or full-closed position indication.

The OBV inlet is flange-mounted to the preburner oxidizer supply duct at the FPOV location and the oxidizer bleed duct is welded to the valve outlet. The FBV inlet is flange-mounted to the fuel high-pressure duct and the fuel bleed duct is welded to the valve outlet.

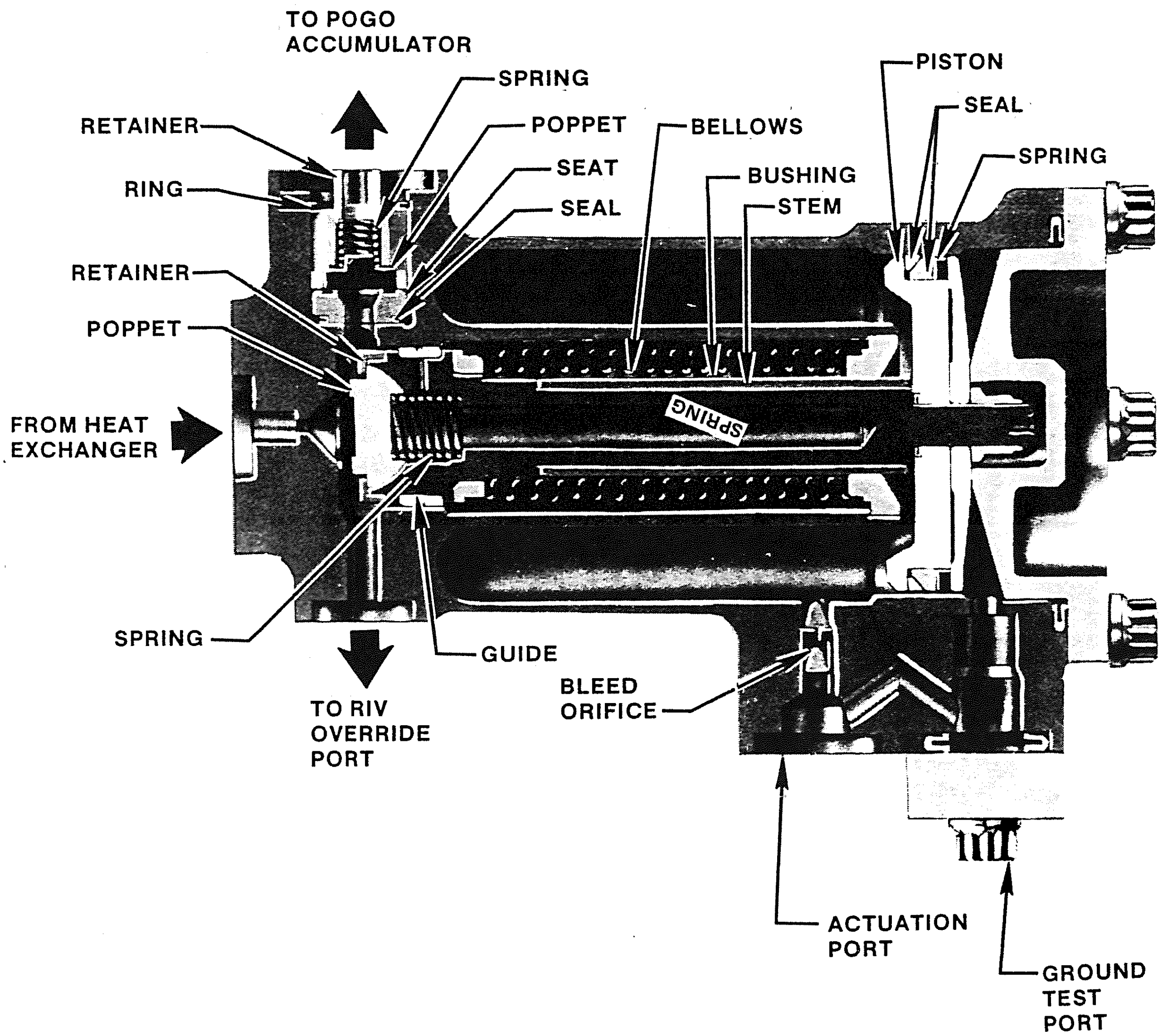


Figure 1.6-8.- GCV

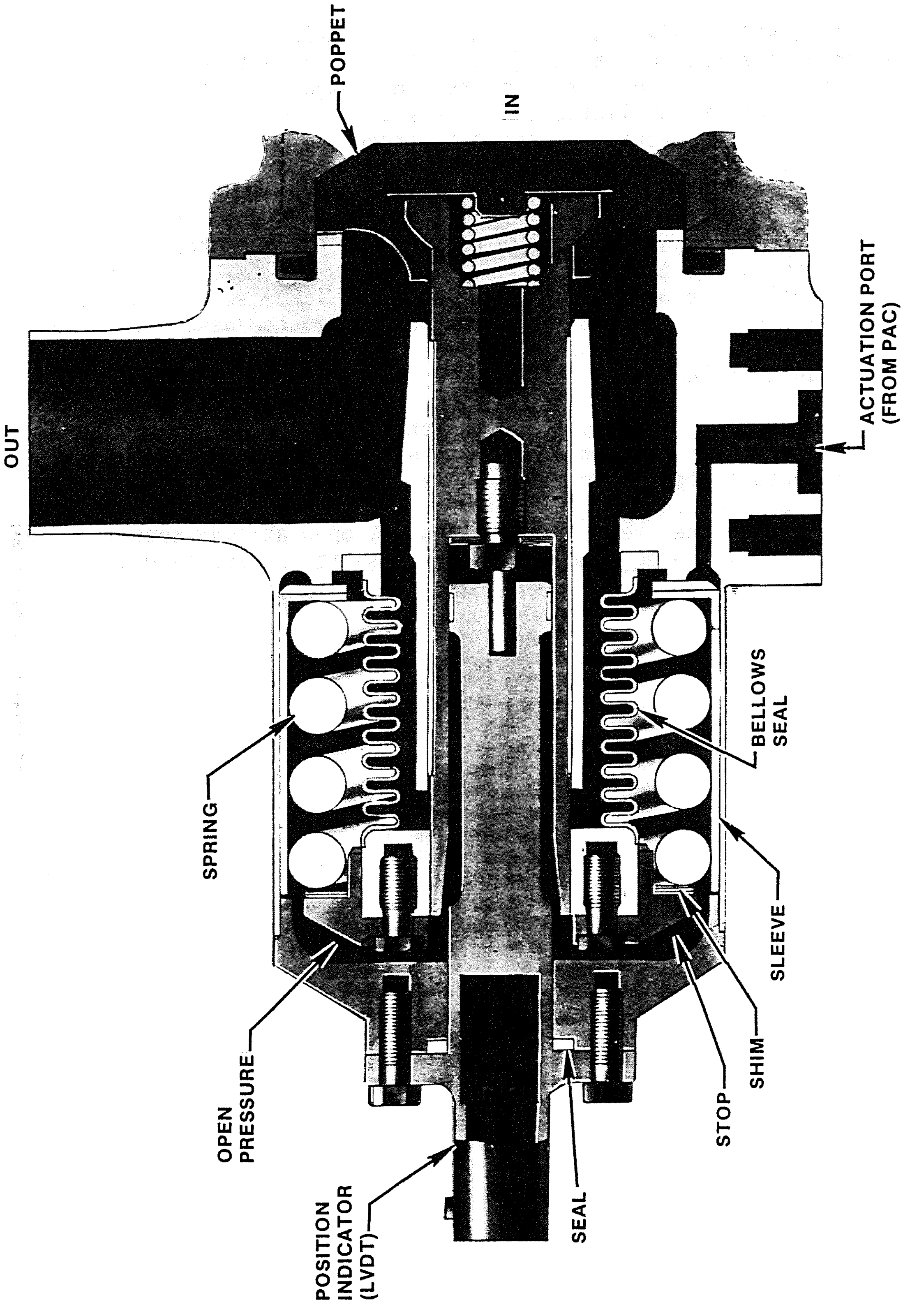


Figure 1.6-9.- OBV and FBV.

1.6.2.10 Heat Exchanger Antiflood Valve

The antiflood valve (AFV) is shown in figure 1.6-10. The AFV prevents LOX from entering the heat exchanger (HEX) until sufficient heat is available in the high-pressure oxidizer turbopump turbine exhaust gas to vaporize the LOX for oxidizer tank pressurization. The AFV also functions as a pressure relief valve in the reverse direction to prevent heat exchanger overpressurization.

The AFV is a poppet-type valve, spring-loaded closed with one dynamic seal to control oxidizer leakage between the piston and valve housing. Leakage past the piston is vented through the oxidizer drain port to maintain the AFV operational. The poppet pivots so it can align itself with the seal. A dual channel position indicator is incorporated in the design to enable controller monitoring of valve position. A ground actuation port is also provided to enable inplace checkout without breaking engine connections.

The AFV is set to open during the start phase. The oxidizer system pressure is an indicator of an adequately heated heat exchanger, and the AFV is adjusted to actuate open when the oxidizer system pressure reaches a predetermined value. The controller monitors the AFV position and initiates engine shutdown in the event the valve is not open at 1.45 seconds after shutdown. The AFV closes when the oxidizer system pressure decays.

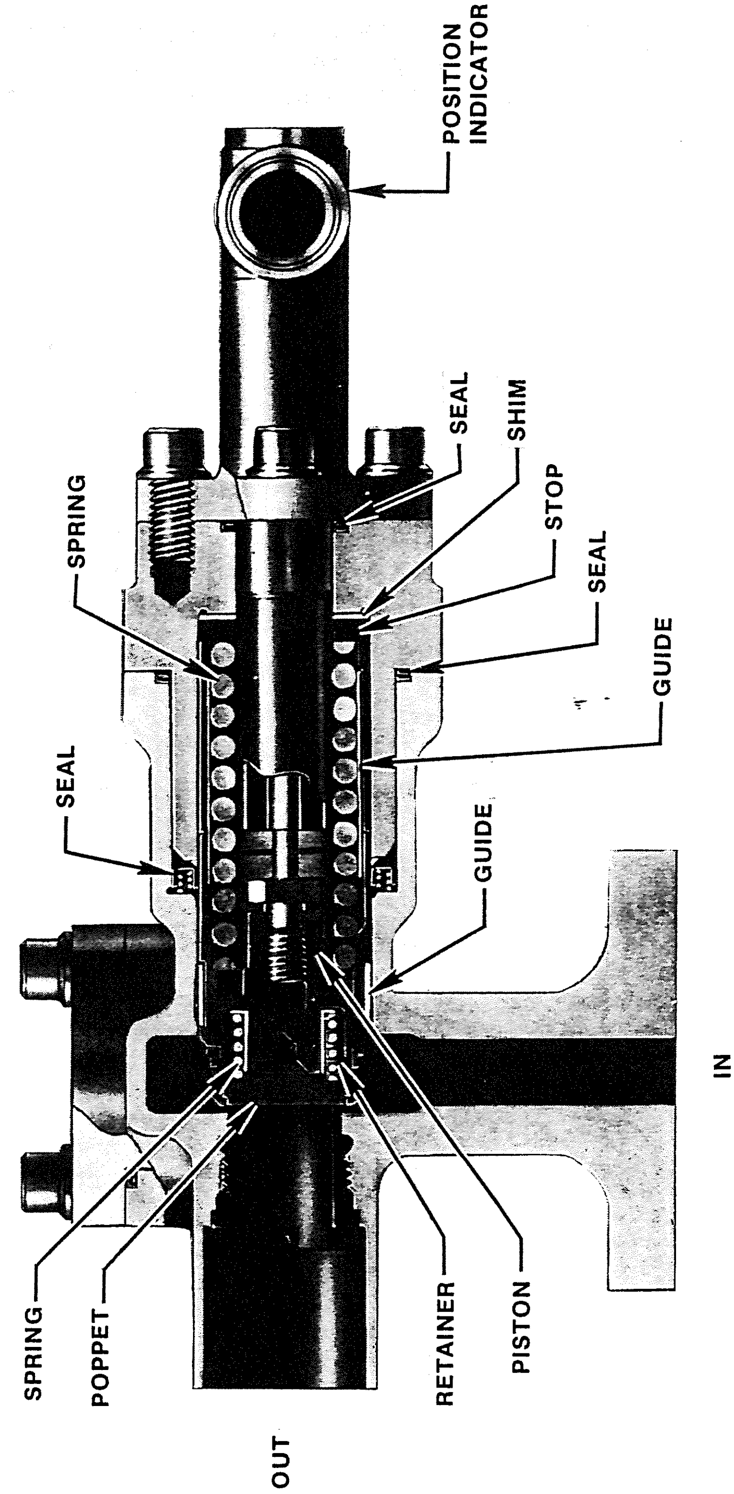
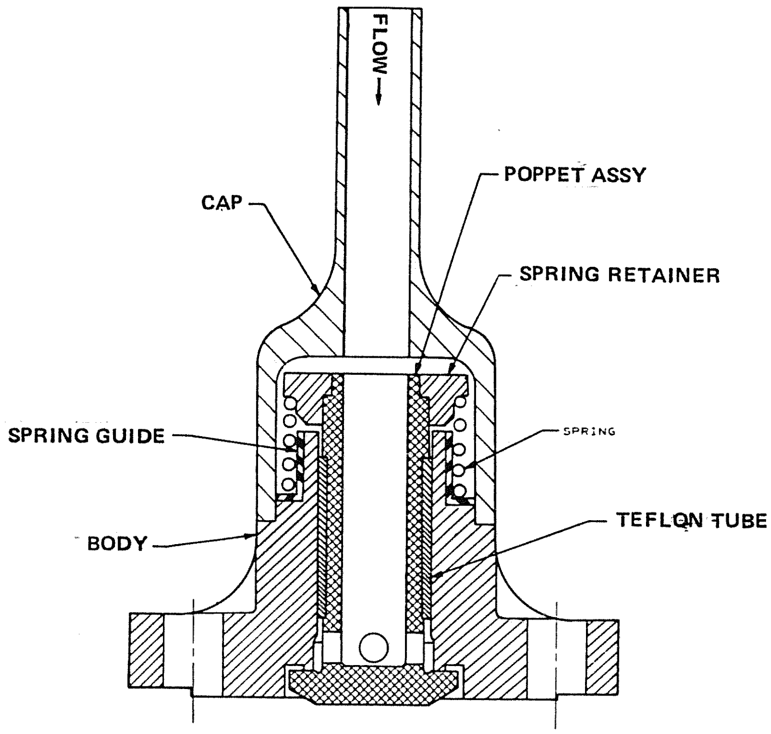


Figure 1.6-10.- AFV

1.6.2.11 Helium Purge Check Valve

A schematic of the helium purge check valve is shown in figure 1.6-11. The pneumatic control system purge check valves are spring-loaded, normally closed, pressure-actuated open poppet valves that isolate propellants from the pneumatic systems.



HELIUM PURGE CHECK VALVE
RS008059

1. OXIDIZER PRE-BURNER OXID. DOME
2. FUEL PRE-BURNER OXID. DOME
3. MAIN COMBUSTION CHAMBER OXID. DOME
4. DOWNSTREAM OF MAIN FUEL VALVE
5. HPOT TURBINE SEAL CAVITY

Figure 1.6-11.- Helium purge check valve.

1.6.2.12 Normally Open Solenoid Valve

The schematic for a normally open solenoid valve is shown in figure 1.6-12. The emergency shutdown solenoid valve is such a valve.

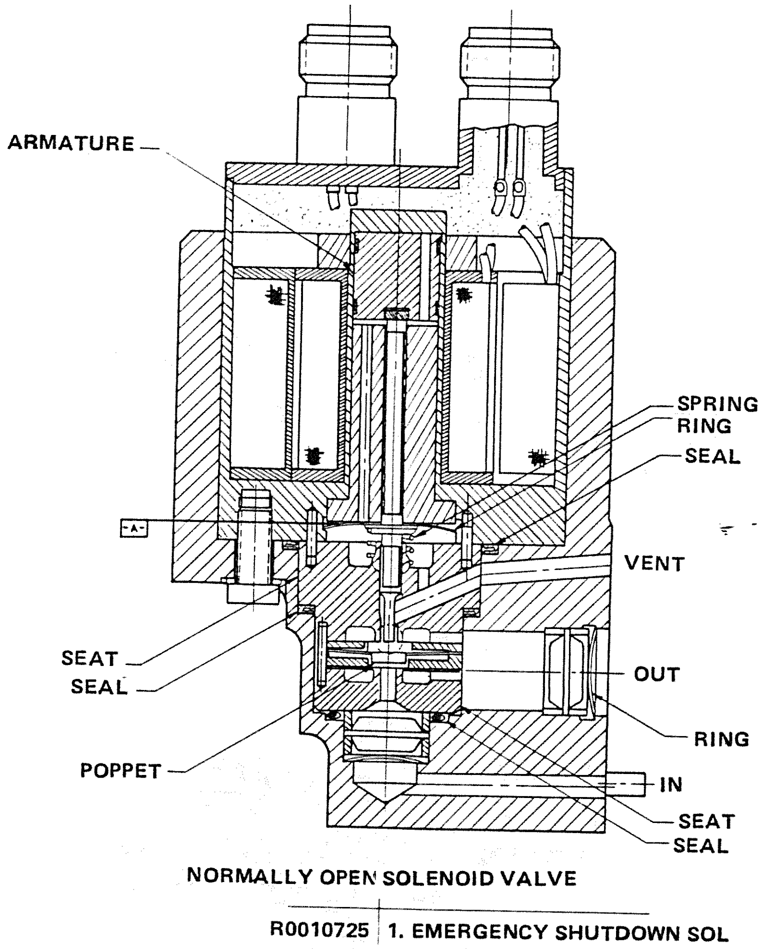


Figure 1.6-12.- Normally open solenoid valve.

1.6.2.13 Normally Closed Solenoid Valve

A schematic of a normally closed solenoid valve is illustrated in figure 1.6-13. The following valves are normally closed solenoid valves: fuel system purge solenoid, bleed valve control solenoid, HPOT intermediate seal purge solenoid, oxidizer preburner dome purge solenoid, and helium precharge valve solenoid.

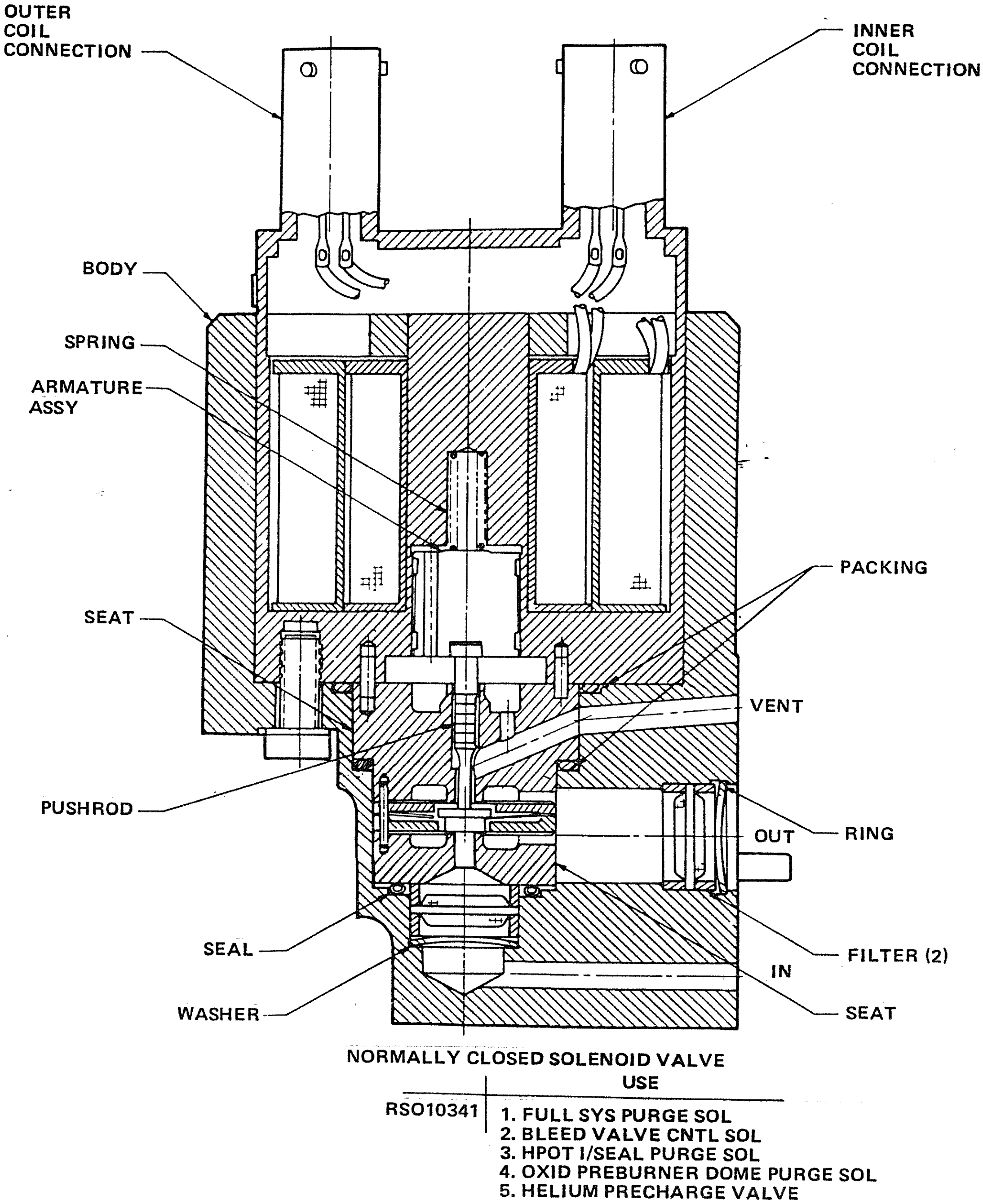
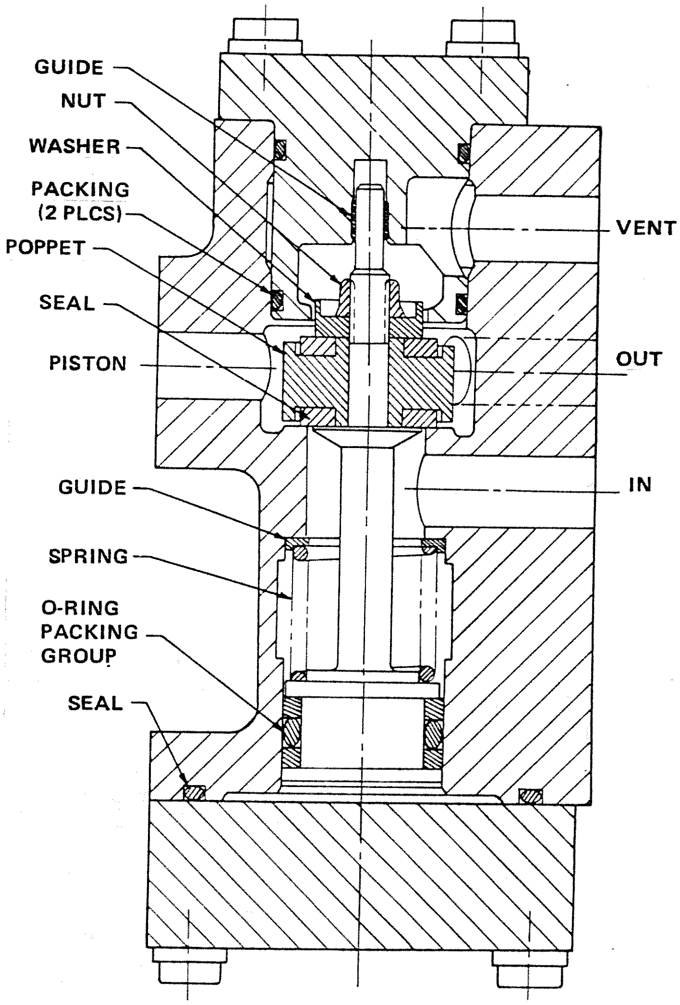


Figure 1.6-13.- Normally closed solenoid valve.

1.6.2.14 Pressure Actuated Valves

The schematics of pressure actuated valves are shown in figure 1.6-14.

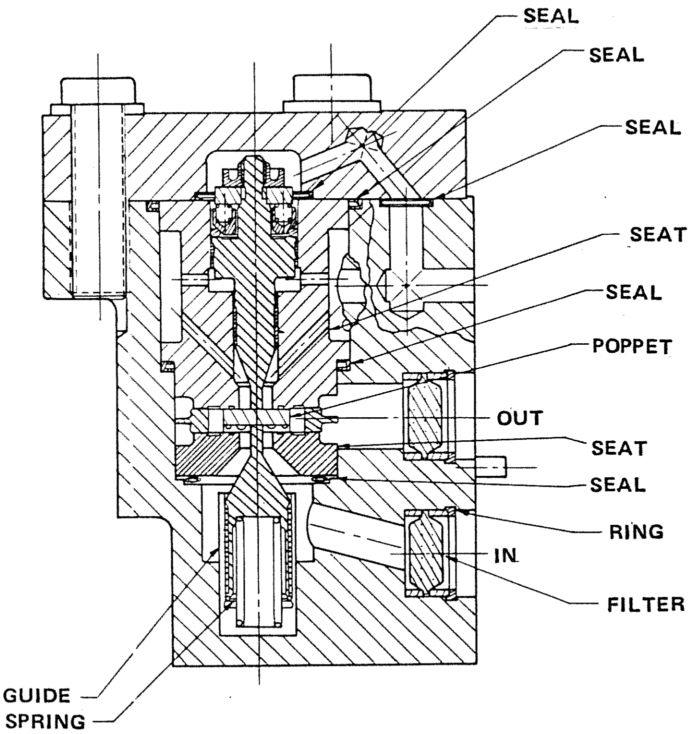


PRESSURE ACTUATED VALVE

RS008021

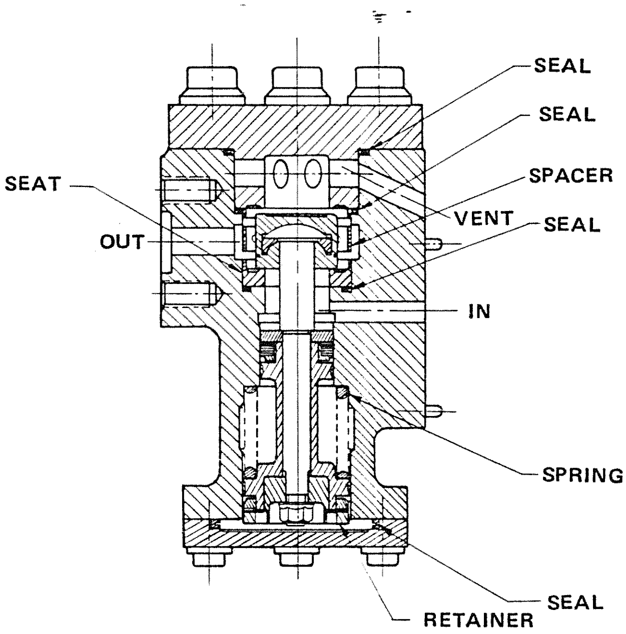
1. EMERGENCY SHUTDOWN CONTROL
2. OXIDIZER BLEED
3. FUEL SYSTEM PURGE
4. OXIDIZER BLEED VALVE
5. OXIDIZER SYSTEM PURGE

Figure 1.6-14.- Pressure-actuated valves (fig. 1 of 3).



PRESSURE ACTIVATED VALVE

R0019401 | 1. PURGE SEQUENCE VALVE



PRESSURE ACTIVATED VALVE

R0010984 | 1. FUEL PREBURNER PURGE

Figure 1.6-14.- Pressure-actuated valves (fig. 2 of 3).

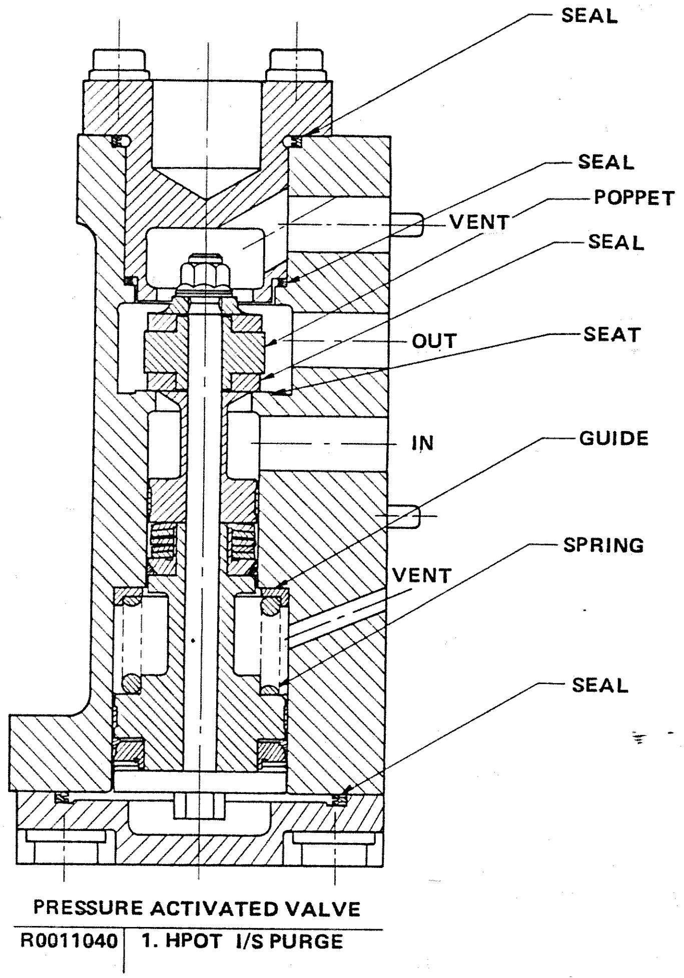


Figure 1.6-14.- Pressure-actuated valves (fig. 3 of 3).

1.6.2.15 Propellant Valve Hydraulic Actuators

A cutaway view of a typical actuator is shown in figure 1.6-15. A schematic of the main propellant valve actuator and a schematic of the preburner valve actuator are included in figure 1.6-16. A schematic of the cross section of the propellant valve actuator is shown in figure 1.6-17.

Hydraulic power is provided for the operation of five valves in the propellant feed system (oxidizer preburner oxidizer, fuel preburner oxidizer, main oxidizer, main fuel, and chamber coolant valves). Servoactuators mounted to the propellant valves convert vehicle-supplied hydraulic fluid pressure to rotary motion of the actuator shaft as a function of electrical input command.

Two servovalves, which are integral with each servoactuator, convert the electrical command signal from the engine controller to hydraulic flow to position the valve actuator. The dual servovalves provide redundancy that permits a single servovalve failure with no change in actuator performance. A fail-operate servoswitch is used to automatically select the redundant servovalve upon failure of a single servovalve. A fail-safe servoswitch is used to hydraulically lock up the servoactuator upon failure of both servovalves.

A heater is installed on the MFV actuator neck (at the valve interface flange) to maintain the hydraulic fluid temperature at the required level.

A dual, redundant, rotary variable differential transformer (RVDT) is connected to the actuator shaft and returns two electrical signals of actuator position to the controller.

All actuators are capable of using an emergency shutdown system to pneumatically close the propellant valves. Pneumatic sequence valves in the oxidizer and FPB actuators provide for a proper closing sequence of the propellant valves during an emergency shutdown condition. A pneumatic sequence valve in the CCV actuator is used to terminate the engine postshutdown purges.

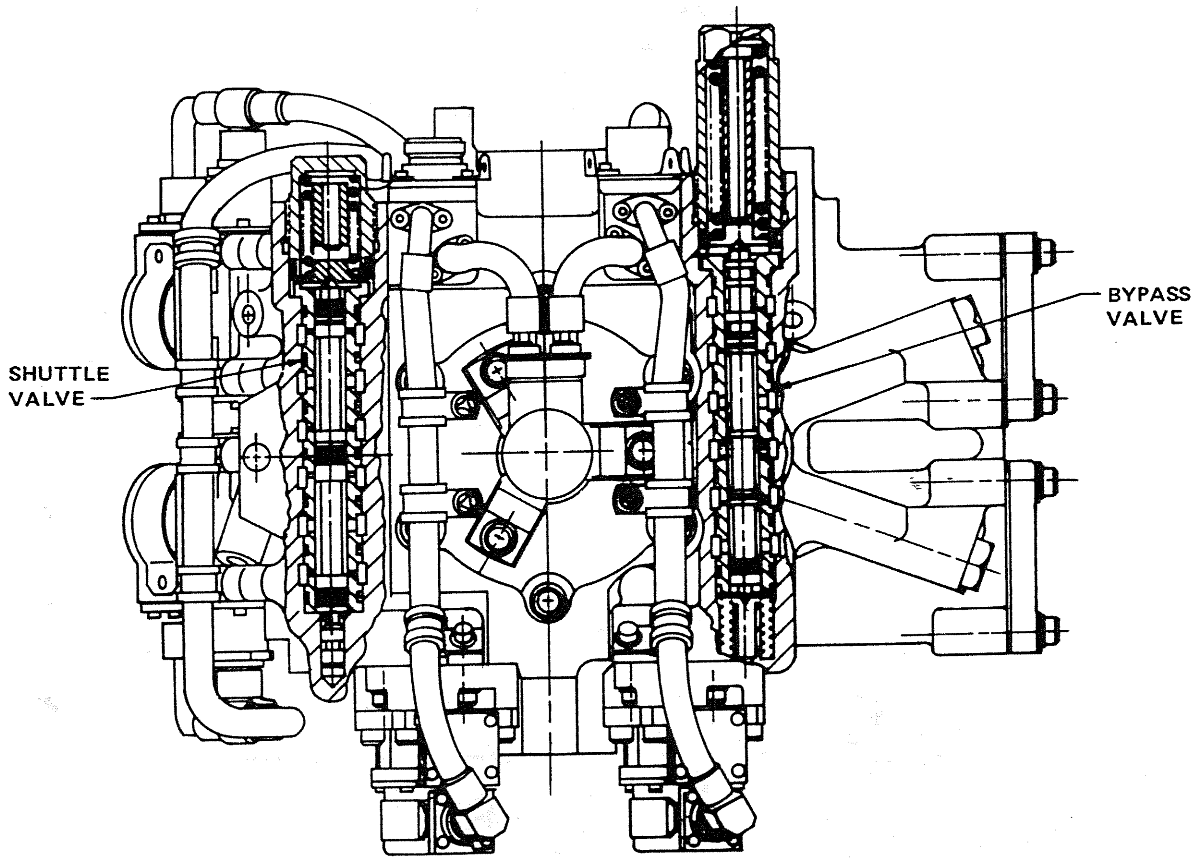
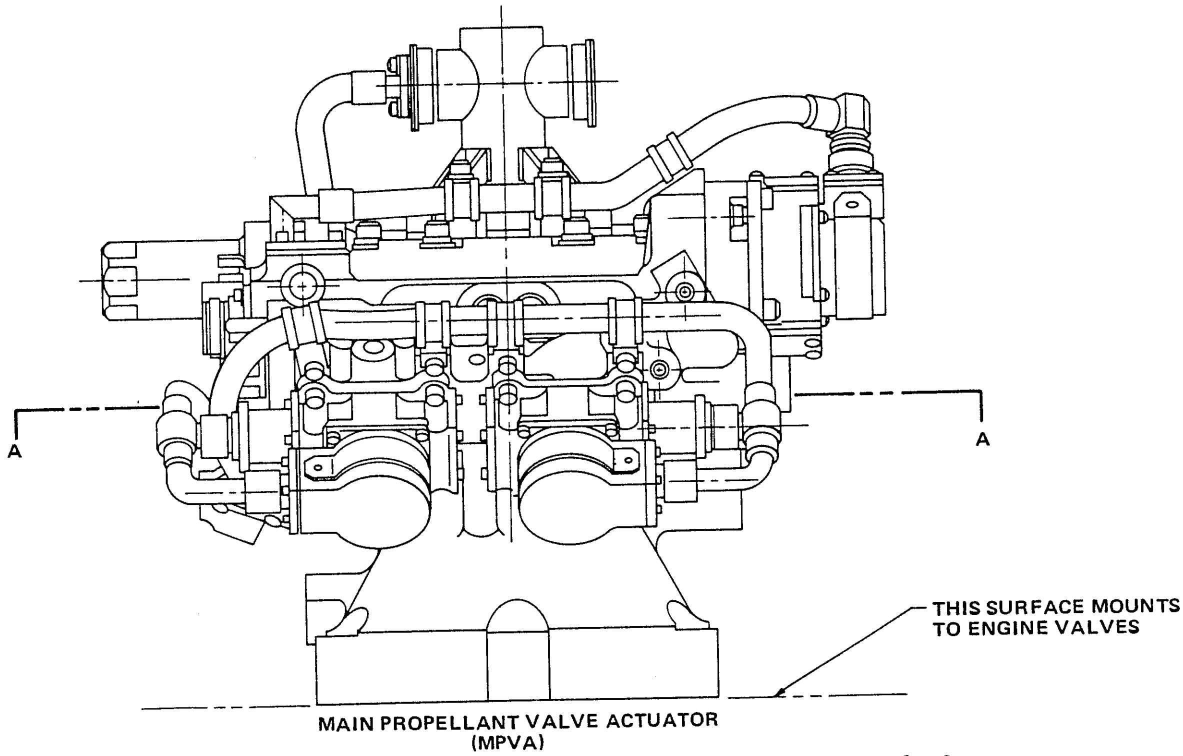


Figure 1.6-15.- Propellant valve actuator cutaway view.



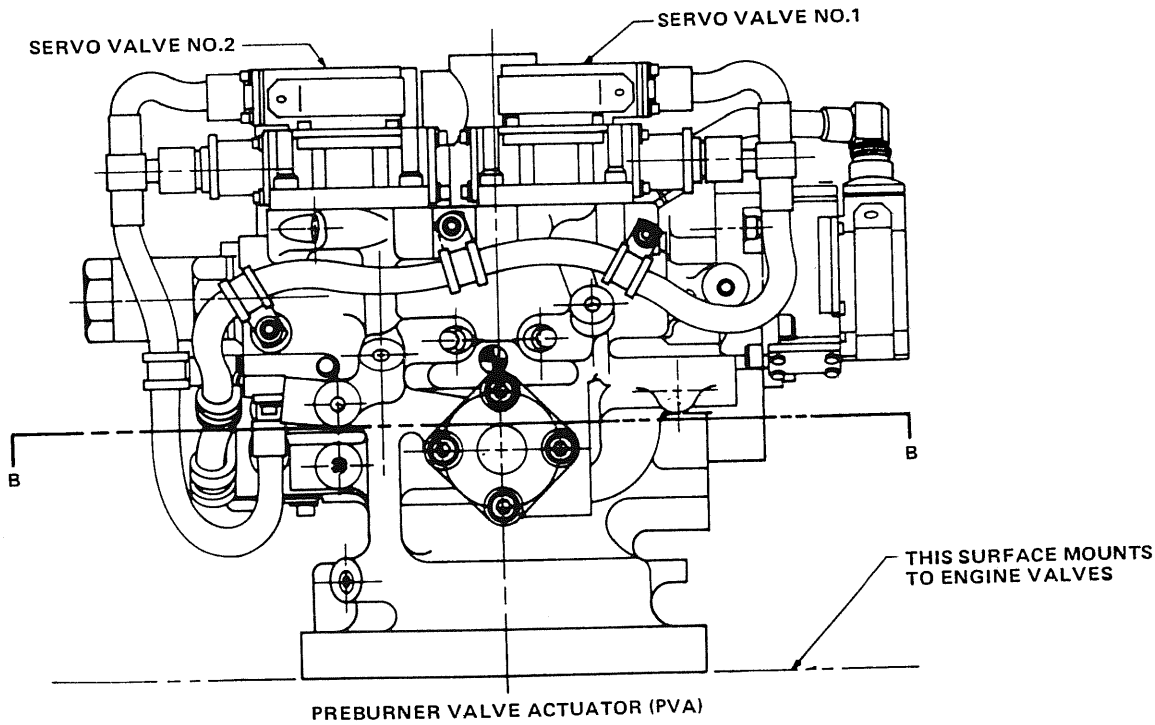


Figure 1.6-16.- Propellant valve actuators.

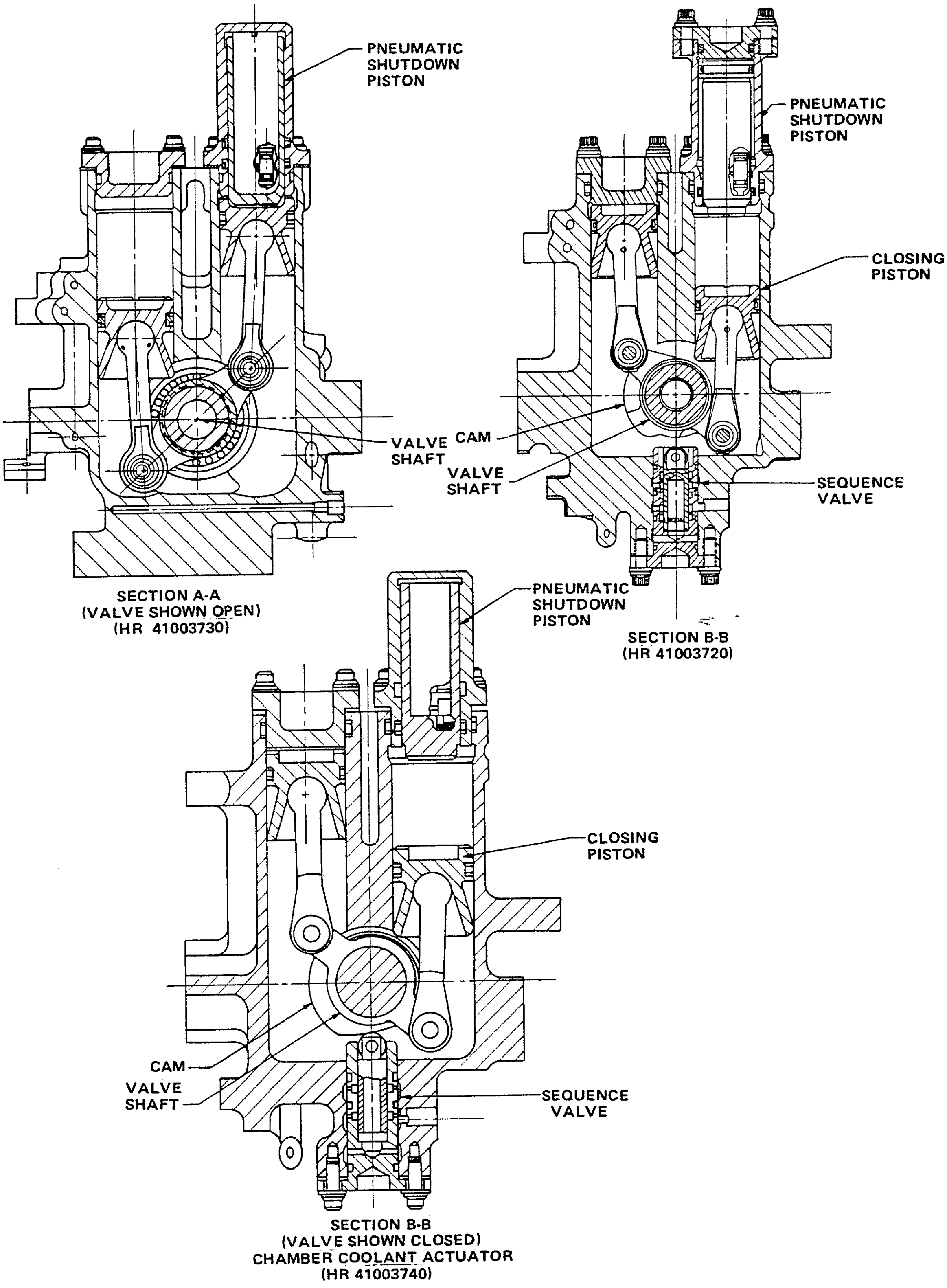


Figure 1.6-17.- Propellant valve actuator cross section.

1.7 SSME PURGE/ACTUATOR OPERATIONS

1.7.1 Hydraulic Actuator Description

Each of the five engine ball valves is controlled by a fail-operational/ fail-safe actuator. The actuators are hydraulically driven and are designed with a pneumatic piston that is used for redundant closing of the valve at engine shutdown. The actuator hardware includes channel A and B servo-valves, a fail-to-operate servoswitch, a shuttle valve, a fail-safe servoswitch, a bypass valve, and a pneumatic sequence valve.

1.7.1.1 Servovalves

Individual servovalves are dedicated to channel A and channel B of the main engine controller (figs. 1.7-1 and 1.7-2). The servovalve controls the movement of the actuator by metering hydraulic flow to either the open or closed side of the actuator piston. The servo pulls a flapper valve toward either an opening or closing orifice, which creates a pressure unbalance across the servopiston and allows pressure to enter either the open or closed side of the actuator. Channel A of the controller is normally in control. The rotational variable differential transformer (RVDT) measures the actuator position as a percent of travel. The position is sent to the output electronics box in the controller where it is demodulated. The position is then sent to a servovalve driver where the actuator position is compared to the commanded position. The measured position is also used in an analog model to detect actuator channel errors. If the error between the commanded position and the measured position on channel A is greater that 6 percent, valve control is switched to channel B by means of the fail-to-operate servoswitch. The servoactuator A failure is reported with an octal failure identification (FID) 015 and an octal delimiter in vehicle data table word 5. The FID indicates that a servoactuator has failed. The delimiter indicates the channel and valve that failed.

1.7.1.2 Fail-to-Operate Servoswitch

When channel A servovalve is in control, the fail-to-operate servoswitch is in the de-energized state. The servoswitch driver in the output electronics energizes the fail-to-operate servoswitch when a 6 percent error is detected on channel A.

The flapper valve is energized to cover the channel B orifice. A pressure activated slide valve allows hydraulic pressure to be channeled to the shuttle valve where a spring-loaded slide valve is moved to the channel B position, thus allowing the channel B servovalve driver to control hydraulic pressure to the actuator (ref. fig. 1.7-3).

1.7.1.3 Fail-Safe Servoswitch

Once the channel B servovalve takes control, monitoring is done by the output electronics to check that the error between the commanded and measured positions is less than 10 percent. The fail-safe servoswitch controls the bypass valve which locks hydraulic pressure simultaneously on the open and closed actuator pistons if both channel A and channel B servovalves fail. While channel B servovalve is in control, the fail-safe servoswitch is in the energized state. This switch maintains a positive pressure on the bypass valve to allow normal hydraulic flow to the open or closed actuator pistons. If the fail-safe servoswitch is de-energized, a flapper valve unbalances a slide valve which then vents holding pressure from the bypass valve (reference fig. 1.7-4).

1.7.1.4 Bypass Valve

The bypass valve will move to the bypass position when both channel A and B servovalves fail or when hydraulic pressure is lost to the actuator. Pressure is simultaneously locked on the open and closed actuator pistons to prevent actuator movement.

1.7.1.5 Hydraulic Lockup

When any of the five engine valves fail the channel A and B checks, the engine will be defined as being in "hard hydraulic lockup." Since it is not desirable for the other engine valves to move when one of the five is locked, the controller software will command all the valves into hydraulic lockup so that affects on mixture ratio and power level can be minimized. The loss of hydraulic pressure from an APU will send all five valves into "soft lockup" if hydraulic pressure drops below 1100 psia. Soft lockup implies that the controller actuator channel checks have not been violated. If no power level commands are being sent, it may take 20 to 60 seconds for the valve to fail the actuator checks. The errors generated by the FPOV and OPOV closed loop control software will be summed until the 6 percent and 10 percent error limits are violated. If hydraulic pressure is somehow miraculously recovered within 20 to 60 seconds, the engine may come out of soft hydraulic lockup and normal engine operation can be resumed. Once the engine is in hard lockup, however, it will stay locked through engine shutdown. The engine design specifications state that the engine must operate if hydraulic pressure is greater that 1500 psia and that it must go into hydraulic lockup below 1100 psia. Actually, lockup may occur at any pressure below 1500 psia. After lockup occurs, the mixture ratio tends to drift upward while chamber pressure tends to drift down. No significant change in high pressure oxidizer turbopump (HPOT) or high pressure fuel turbopump (HPFTP) discharge temperature should occur since both the FPOV and OPOV tend to drift downward. Figure 1.7-5 shows the result of hydraulic lockup testing which occurred from late 1979 through 1981. One lockup occurred during the flight of STS-3 when APU 3 was manually shut down due to a high return lube oil temperature. The right engine (2005) ran stable at

82 percent power level through main engine cutoff (MECO) (22 seconds) when it shut down pneumatically.

1.7.1.6 Pneumatic Shutdown

An engine in hydraulic lockup can no longer be throttled or shut down hydraulically. The engines are designed with a pneumatic control assembly that includes an emergency shutdown solenoid. This solenoid valve is redundantly powered. When shutdown is commanded (at MECO or because of redline exceedance) or when ac power is removed from the controller, the emergency shutdown solenoid is de-energized. Helium pressure at approximately 750 psi is channeled to the bypass valve and to the pneumatic shutdown piston. The bypass valve is moved against a spring force to a position that vents any remaining hydraulic pressure from the open and closed pistons. The pneumatic piston pushes against the closed hydraulic piston and forces the valve closed. On the OPOV and FPOV a pneumatic sequence valve mechanically sequences helium through the five engine ball valves in order to simulate the hydraulic shutdown valve sequence. These two sequence valves are controlled by cams on the OPOV and FPOV actuator shafts. The OPOV routes pneumatic pressure to the MOV and FPOV. After the FPOV moves, pneumatic pressure is routed to close the MFV and the CCV. Figure 1.7-6 shows how pneumatic helium closes the valves. Figures 1.7-8(k) and 1.7-8(l) show how helium is routed for a pneumatic shutdown with and without engine purges.

1.7.2 SSME Pneumatic Control Assembly

The previously discussed pneumatic shutdown is not the only function of the pneumatic control assembly (fig. 1.7-7). This piece of hardware uses five solenoid valves to operate eight pressure activated valves (PAV's), which control pneumatic helium flow for engine purges and SSME shutdown. A dedicated engine helium supply system sends 750 psi regulated pressure to each engine. Figures 1.7-8(a) through 1.7-8(l) show how helium is used during prelaunch purges, in-flight purges, and shutdown activation and purges. Figure 1.7-8(b) also shows how ground-supplied nitrogen is used for engine postflight maintenance. A brief discussion of the engine purges is included in this systems brief.

1.7.2.1 Ground Controlled GN_2 Purge

During ground maintenance and turnaround operation, a GN_2 drying purge is supplied to the main chamber oxidizer dome, HPOTP seal package, OPOV, FPOV, HPOT and HPFT preburner domes. Figure 1.7-8(b) shows how this purge is accomplished. This purge also maintains a positive inerting purge pressure that prevents ingestion of dust or water vapor between flights.

1.7.2.2 Fuel System Purge

Since nitrogen will freeze when it contacts cryogenic hydrogen, a 125-scfm helium purge is used to clear nitrogen from the high pressure engine passages. The purge occurs during purge sequences 2, 3, and 4 and is introduced downstream of the MFV. Figure 1.7-8(c) shows how this purge is accomplished. Frozen nitrogen can cause blockage of engine coolant passages and injector tubes which could result in engine damage during engine start and main stage operations.

1.7.2.3 Bleed Valve Activation and Component Fuel Drain

Bleed valve activation and component fuel drain occur during purge sequences 3 and 4. Cavities in the LPFT, HPFTP and CCV are drained via the component fuel drain line. These cavities are purged with a 2-scfm helium flow in preparation for engine start. The bleed valve activation allows hydrogen and oxygen to bleed through the engine for propellant conditioning. The LO_2 bleed valve is used to drain back the LO_2 tank to the prescribed flight mass starting at lift-off minus 5 minutes. At the START ENABLE signal, the bleed valves close and the POGO precharge purge is initiated. The bleed valves open again at post-shutdown standby and remain open for the duration of the propellant dump sequence. The LH_2 bleed valve is used to provide a dump path between the LH_2 manifold and the fill and drain line for the LH_2 dump (ref. fig. 1.7-8(d)).

1.7.2.4 HPOT Intermediate Seal Package Purge

During purge sequence 4 and main stage, a 260-scfm helium purge is activated via the HPOTP intermediate seal purge solenoid valve and a pressure activated valve. This purge is required for safe engine operation since it separates hot turbine gas from LO_2 in the HPOT. Engine shutdown is automatically commanded by the controller if this purge is lost. Redline monitoring is done on the intermediate seal (shutdown is commanded if pressure is less than 170 psia) and the secondary seal (shutdown is commanded if the pressure is greater than 100 psia). Operation of this purge is shown in figure 1.7-8(e).

1.7.2.5 Preburner Shutdown Purge

During shutdown the fuel preburner and oxidizer preburner LO_2 domes are purged to prevent H_2 and O_2 ignition in the O_2 dome. An ignition in a dome cavity can result in engine damage. Figure 1.7-8(f) shows how these preburner purges are accomplished during a normal shutdown.

1.7.3 SSME Start Preparation Phase Modes

System purges and propellant conditioning are performed in preparation for engine start. Figures 1.7-8(g) through (j) show how helium is routed for purge sequences 1, 2, 3, and 4. Figures 1.7-9(a) through (d) illustrate the valve timing required for these purges. Figure 1.7-10 shows the helium requirements for purges 1, 2, 3, and 4, main stage, and post-MECO operations. Following is a brief discussion of the prelaunch purges and SSME start phase.

1.7.3.1 Purge Sequence 1 Mode

Functions include verification of propellant valve positions. During this sequence, facility initiates oxidizer system and HPOTP intermediate seal gaseous nitrogen purges.

1.7.3.2 Purge Sequence 2 Mode

Functions include onboard helium purge of fuel system and continuation of purges initiated in purge sequence 1.

1.7.3.3 Purge Sequence 3 Mode

Functions include propellant recirculation (bleed valve operation) and discontinuance of fuel system purge. Note that during purge sequence 3, fuel system purge will be reinstated for 3 minutes each 60-minute period. GN_2 purges initiated in purge sequence No. 1 are continued.

1.7.3.4 Purge Sequence 4 Mode

Functions include fuel system and HPOTP intermediate seal helium purges. All fail-safe solenoid valves are energized. Small valve movement (-0.5 to -1.0 percent) towards the closed position should be seen on the MFV, MOV, FPOV, and OPOV. GN_2 purges initiated in purge sequence No. 1 are continued.

1.7.3.5 Engine Ready Mode

Ready stage of start preparation in which proper engine thermal conditions for start have been attained and other criteria for start have been satisfied. Functions include a continuation of purge sequence 4 function.

1.7.3.6 Start Initiation Mode

Initial functions associated with start sequence are in progress. These include all functions prior to ignition confirmed, at 2300 msec. All purges off and verified. Bleed valves closed and verified. Igniters energized and verified. Thrust control loop is closed.

1.7.3.7 Thrust Buildup Mode

Ignition has been detected by monitoring main combustion chamber pressure, and closed-loop thrust buildup sequencing is in progress. Mixture ratio control loop is closed. POGO suppression accumulator is precharged with helium for 2 seconds. MFV, MOV, and CCV are scheduled.

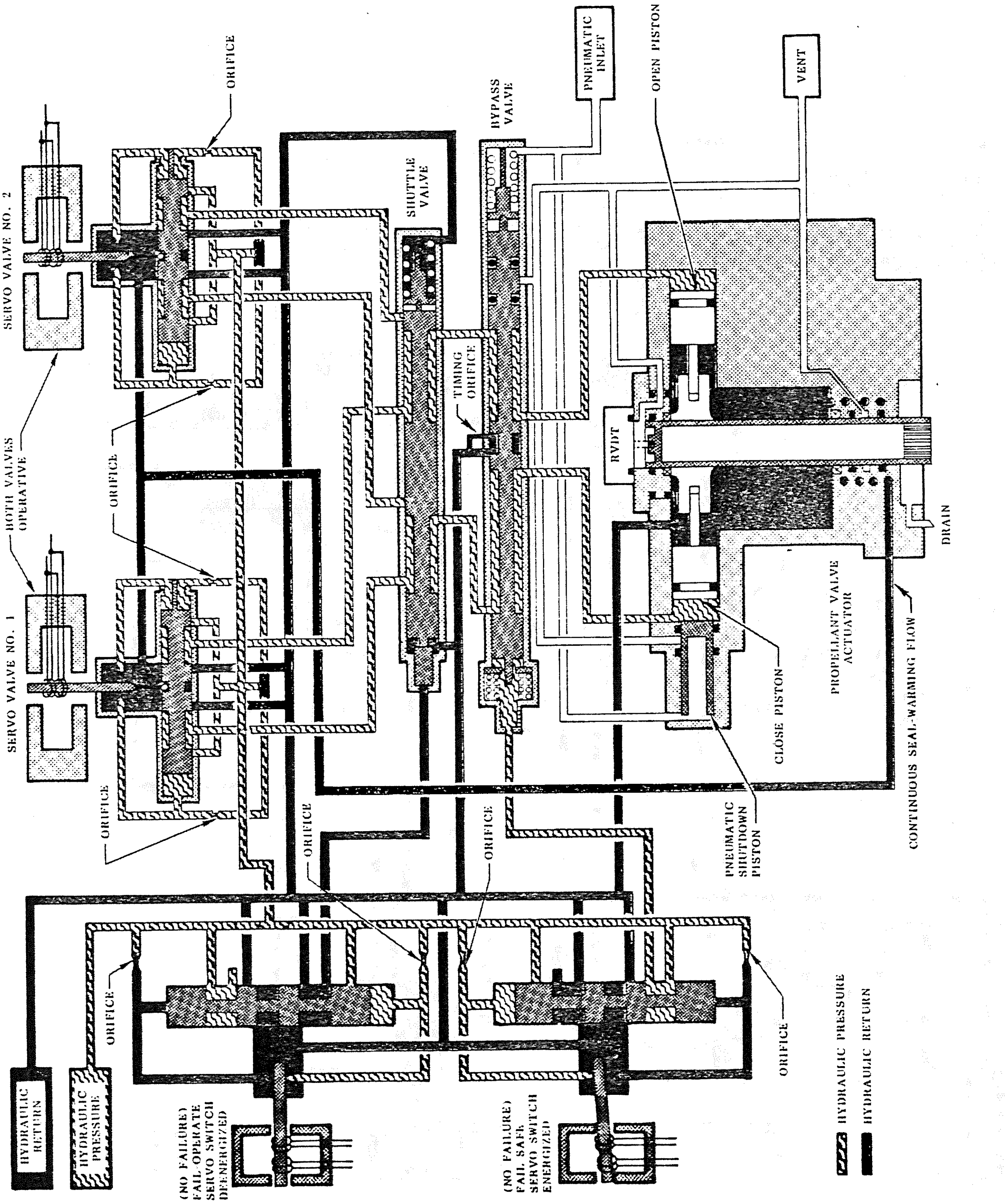


Figure 1.7-1.- Main fuel and main oxidizer valve servoactuator (normal operation).

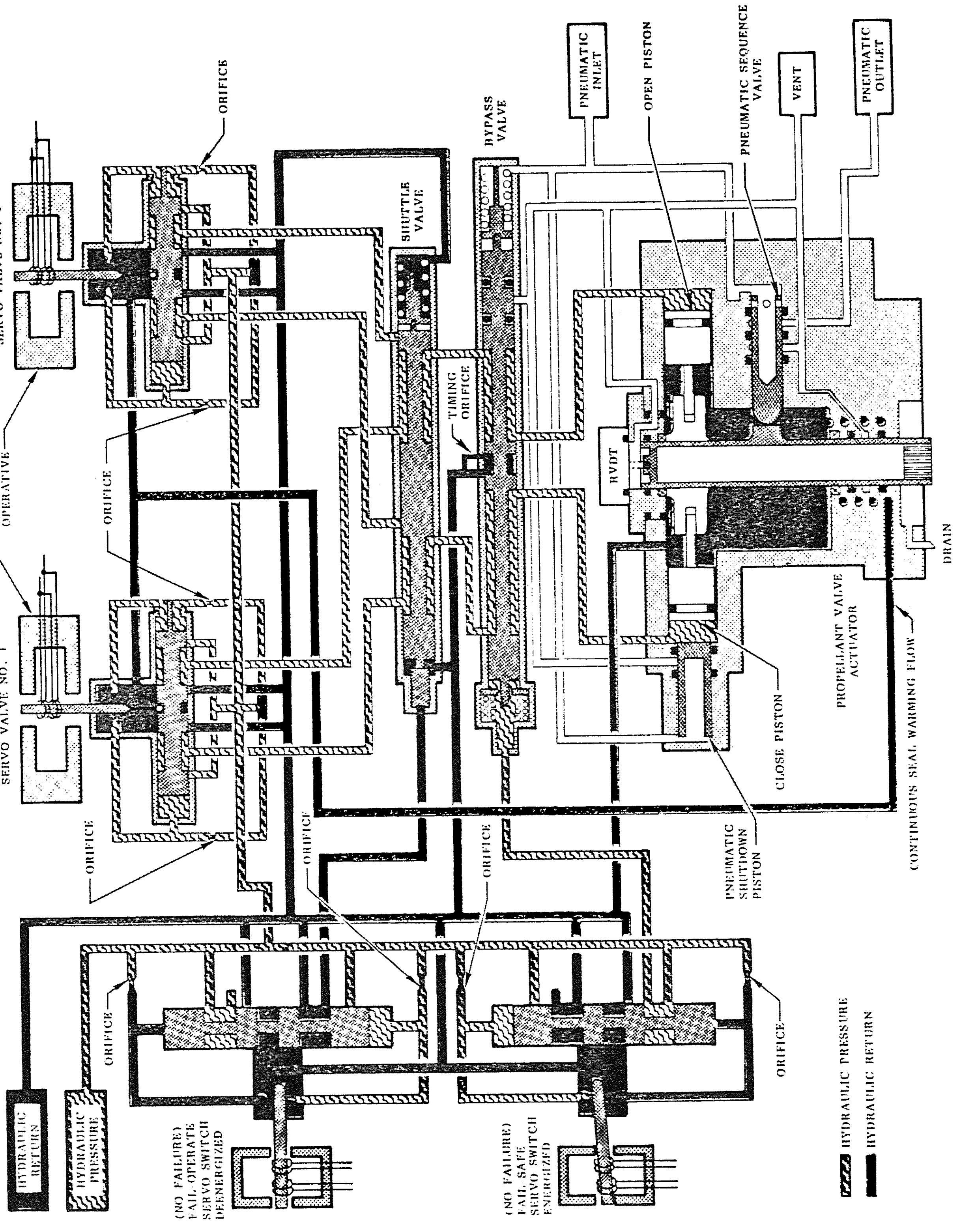


Figure 1.7-2.- Preburner and CCV valve servoactuator (normal operation).

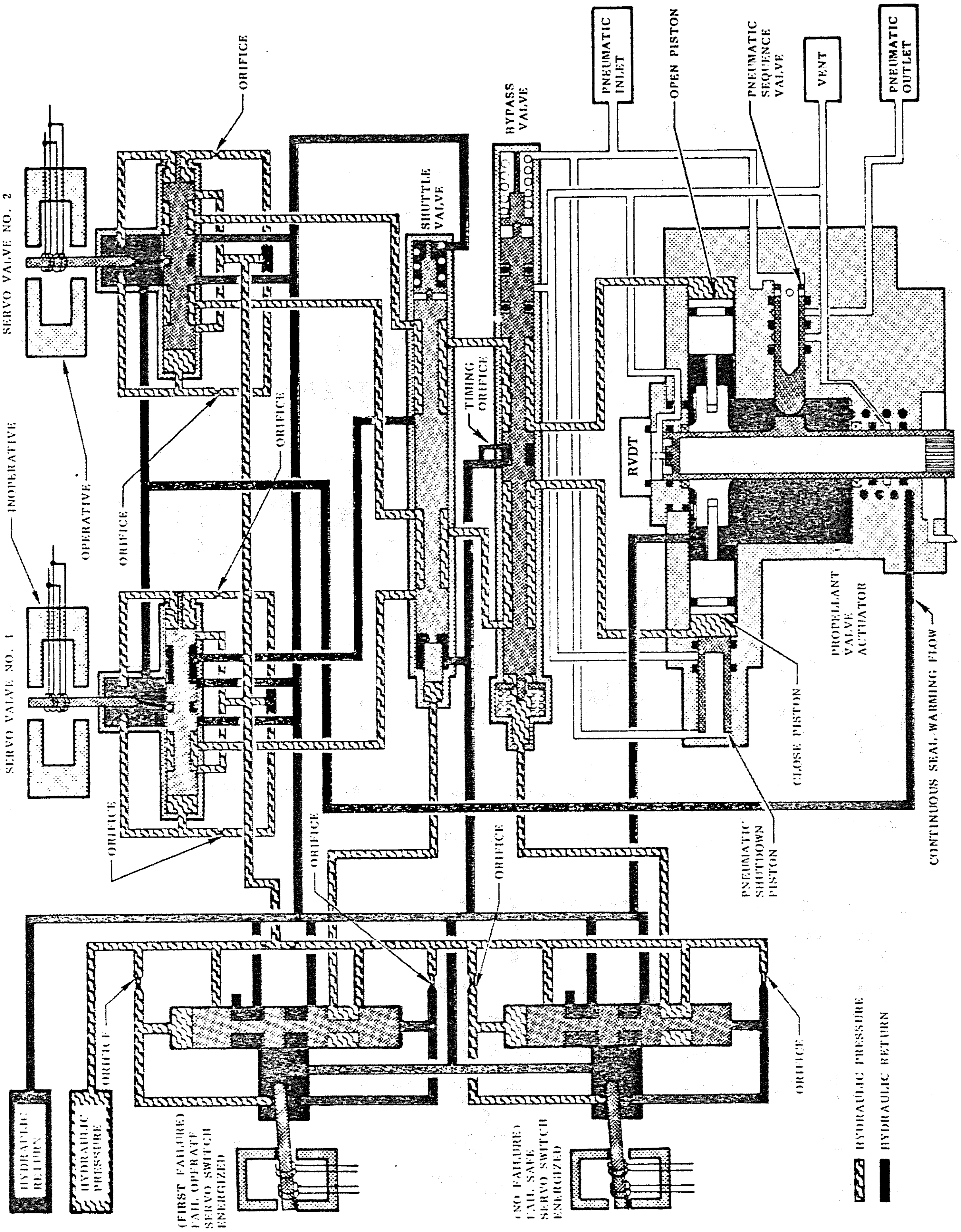


Figure 1.7-3.- Preburner and CCV valve servoactuator (fail-operate configuration).

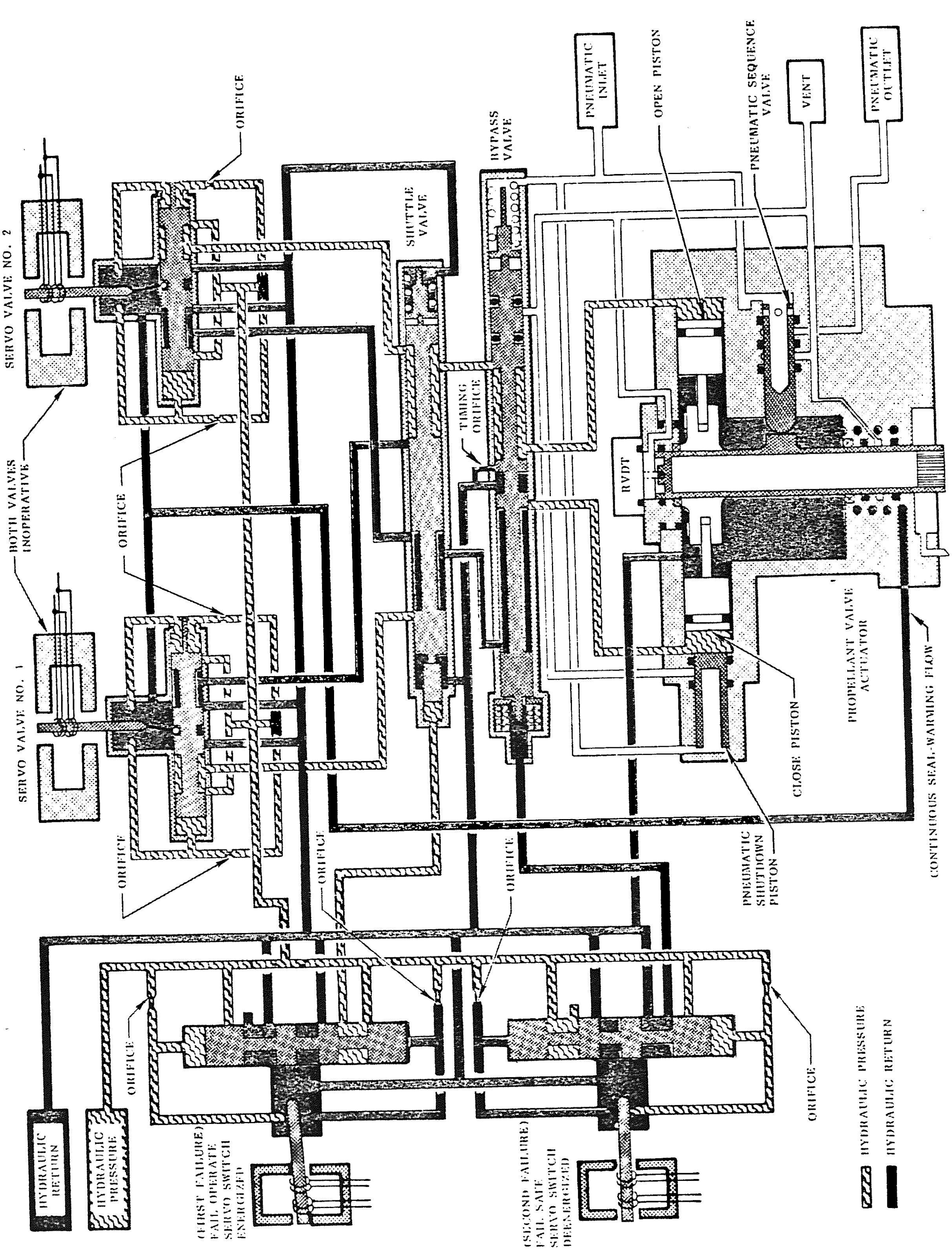


Figure 1.7-4.- Preburner and CCV valve servoactuator (fail-safe configuration).

Engine	Date	Lockup power level	Time in lockup, sec	Mix ratio shift %	PC drift %
0105	Oct '79	100	255	+2.50	-1.2
0007	Nov '80	65	420	-0.78	-2.8*
0007	Dec '80	100	170	+0.32	-0.8
0007	Feb '81	91 (Up-throttle)	10	-3.33	-
2005	STS-3	82 (3g throttle)	22	-	-
2106	May '87	104	190	+0.25	-0.5
2105	May '87	104	190	+2.50	-1.3
2105	June '87	104	190	+0.33	-0.4
2105	June '87	104	636	+11.9	-4.0
2106	June '87	104	107	0	-1.3
2106	July '87	104	184	0	-
0211	Apr '88	104	500	+5.32	-3.0
0211	June '88	104	500	+4.33	-2.7
0211	Oct '88	104	280	+1.25	+1.2

*Percent of RPL P_C.

Figure 1.7-5.- Hydraulic lockup testing.

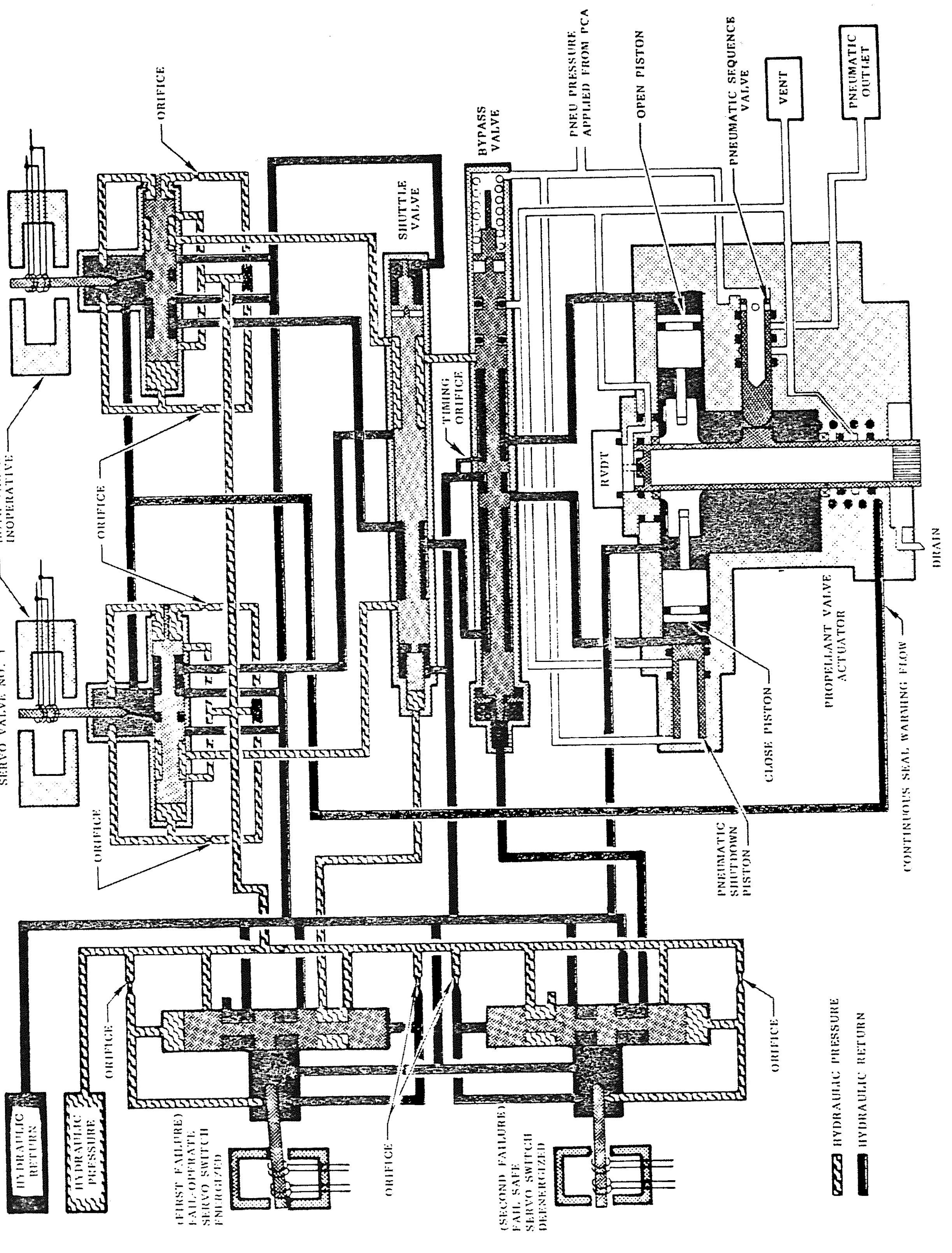


Figure 1.7-6.- Preburner and CCV valve servoactuator (pneumatic shutdown configuration).

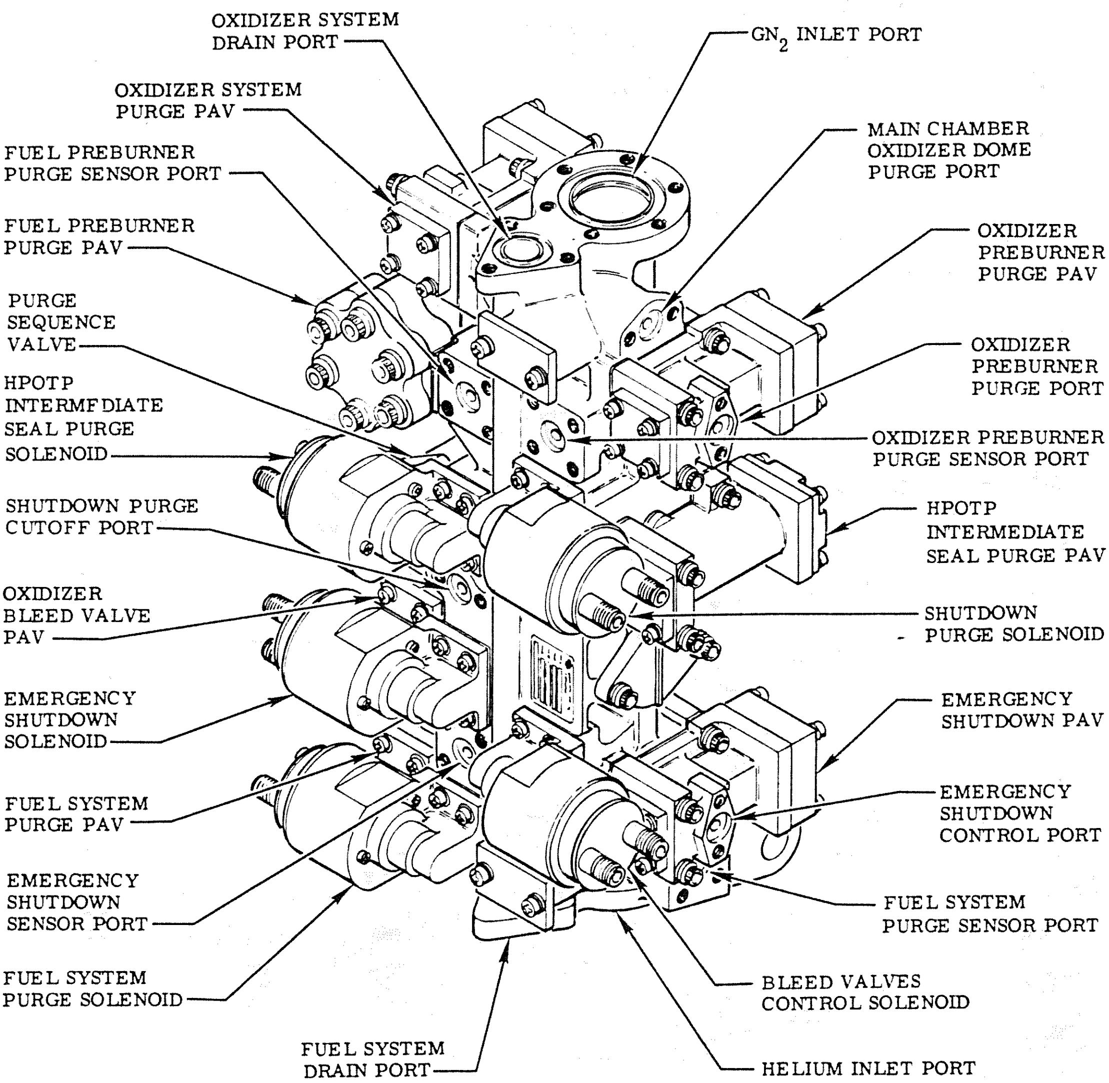
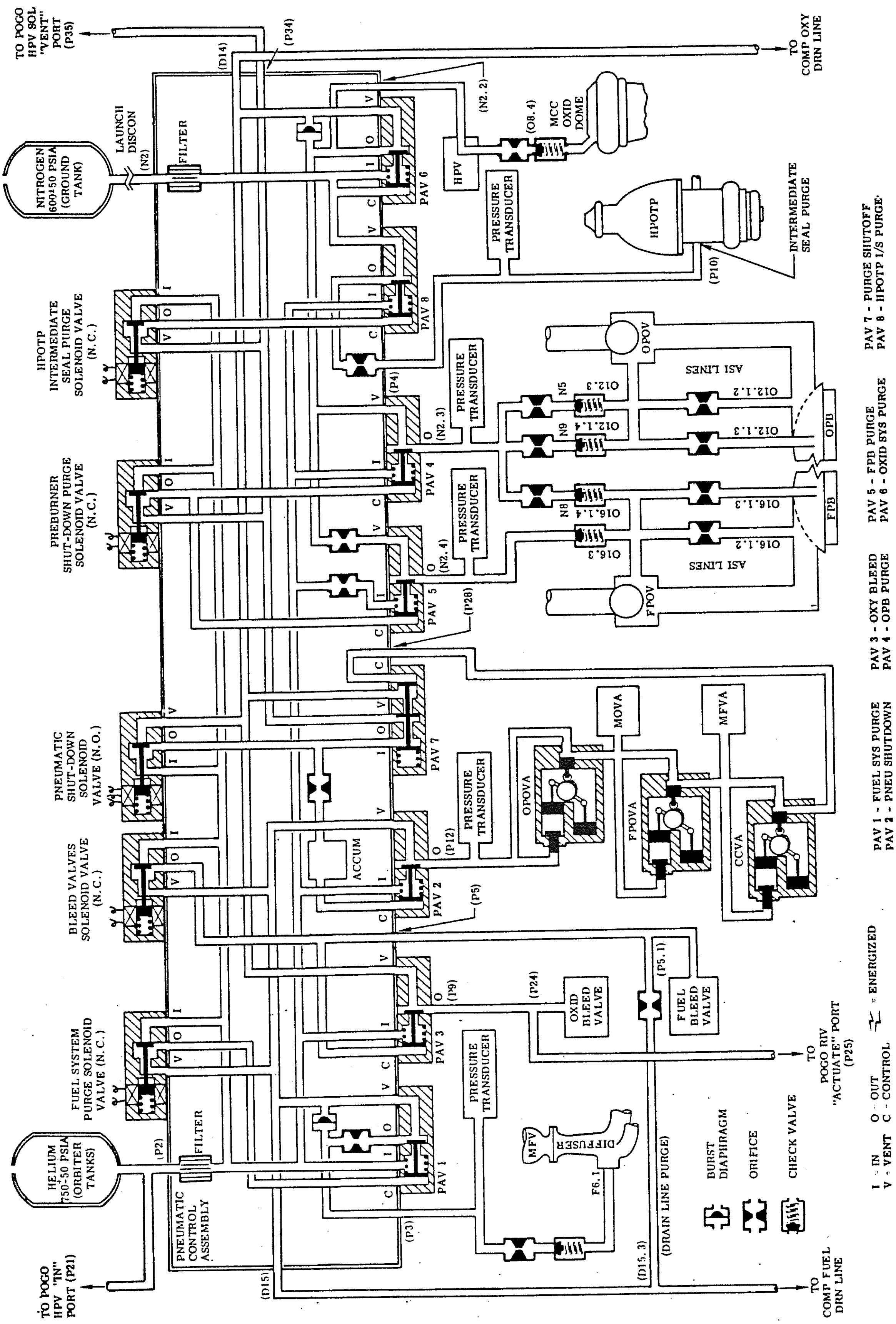
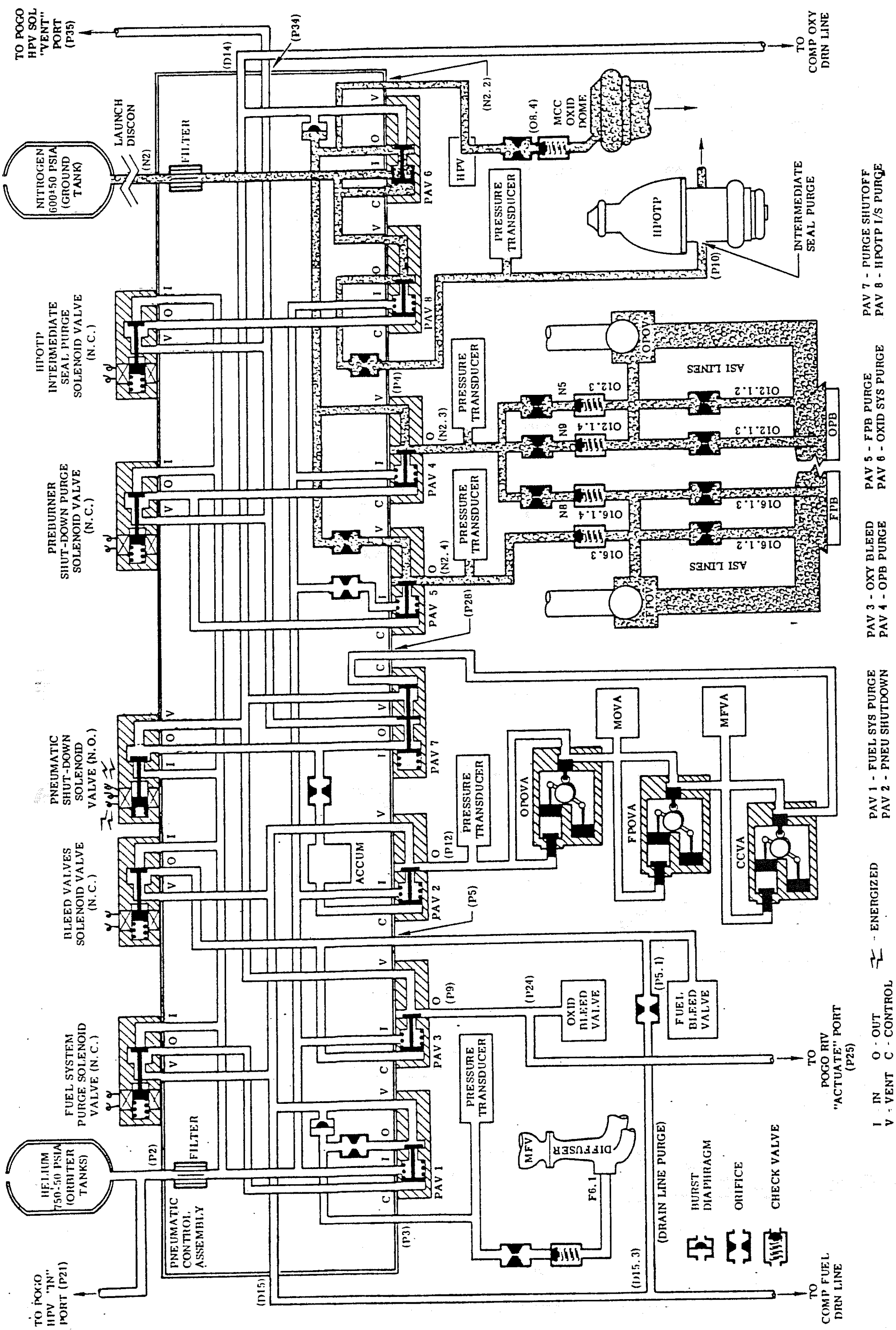


Figure 1.7-7.- Pneumatic control assembly flight configuration.



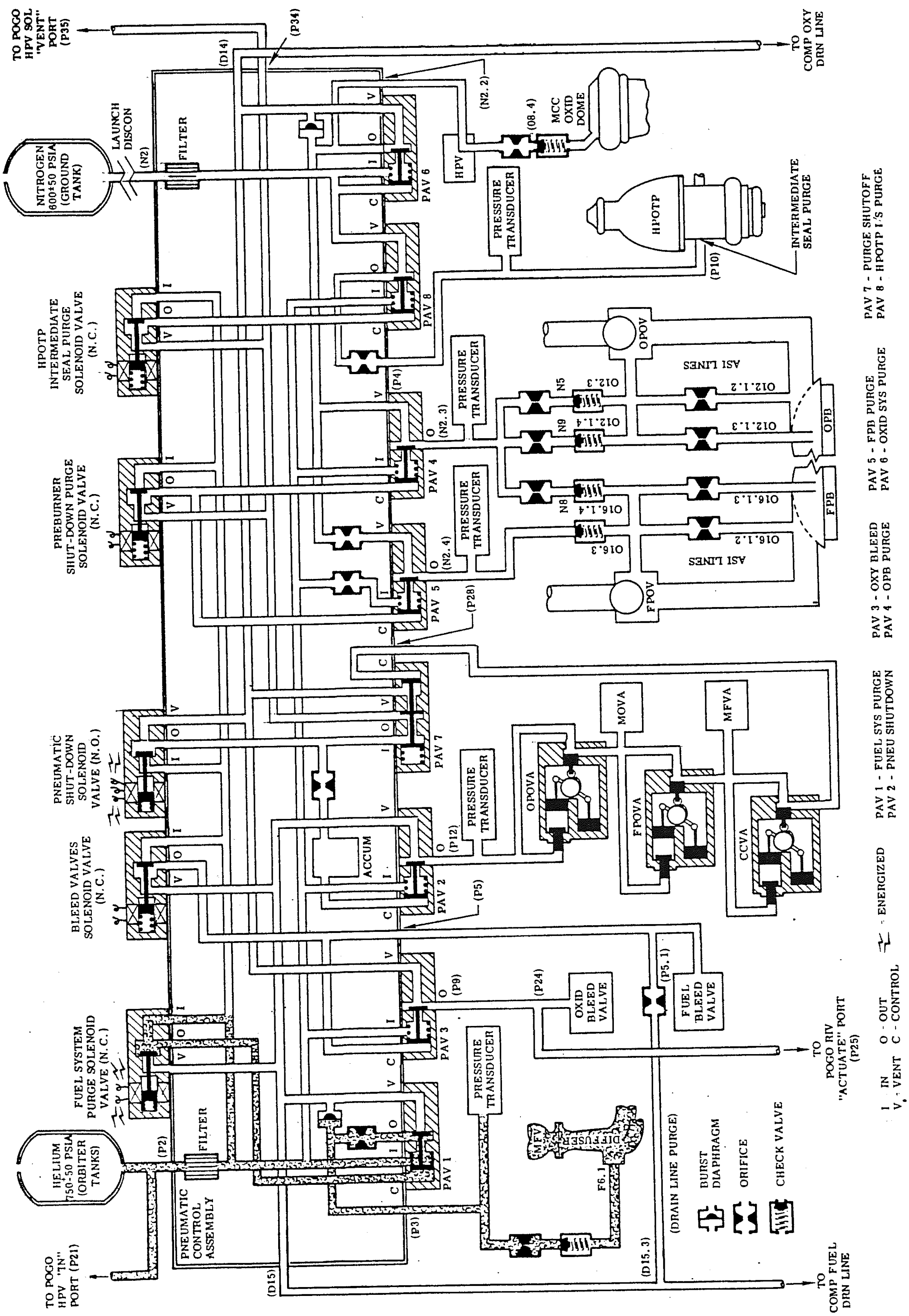
(a) No power-no pressure applied

Figure 1.7- 8.- SSME pneumatic schematic (1 of 12).



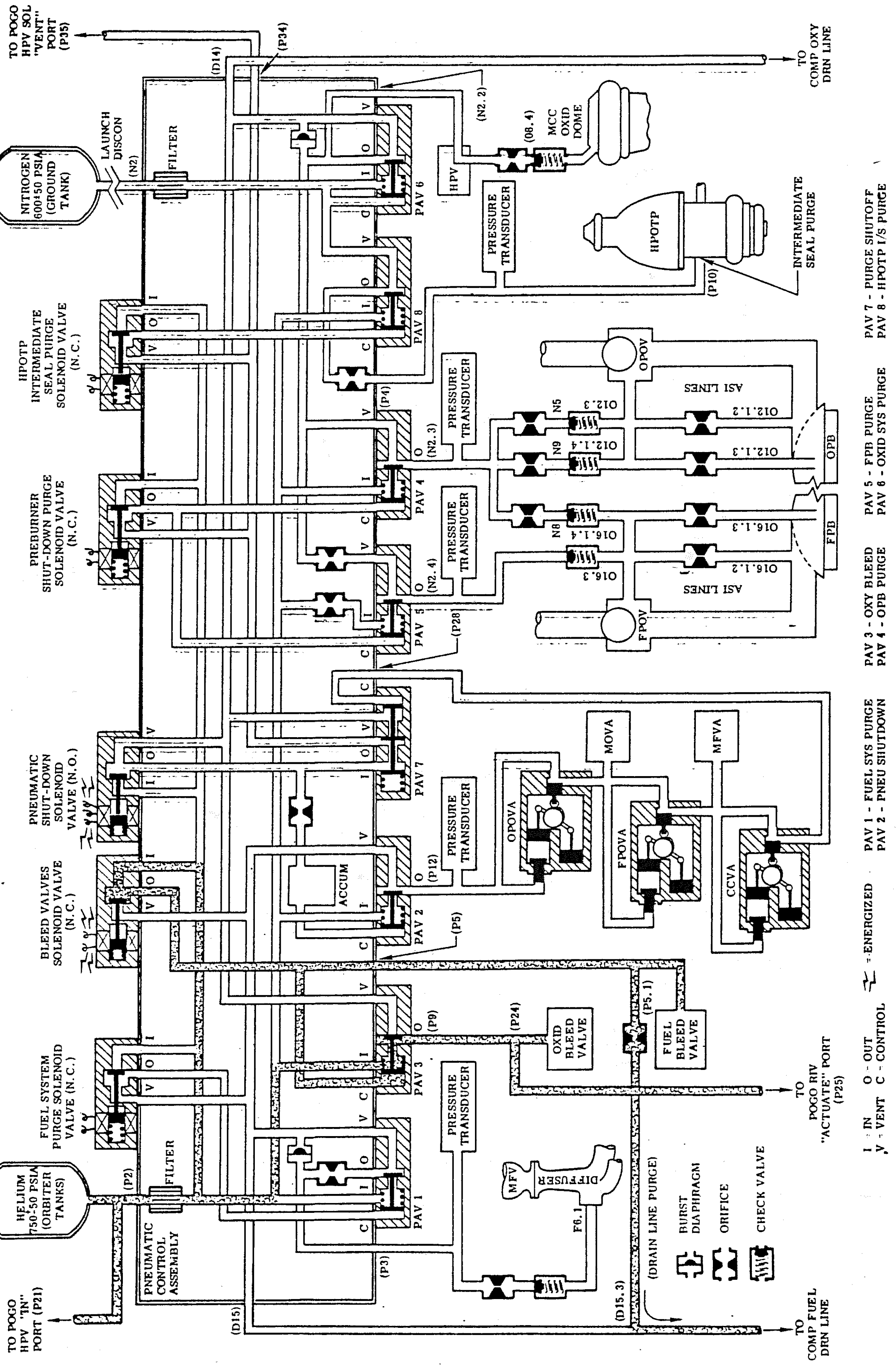
(b) Ground controlled GN_2 purge.

Figure 1.7-8.- Continued (2 of 12).



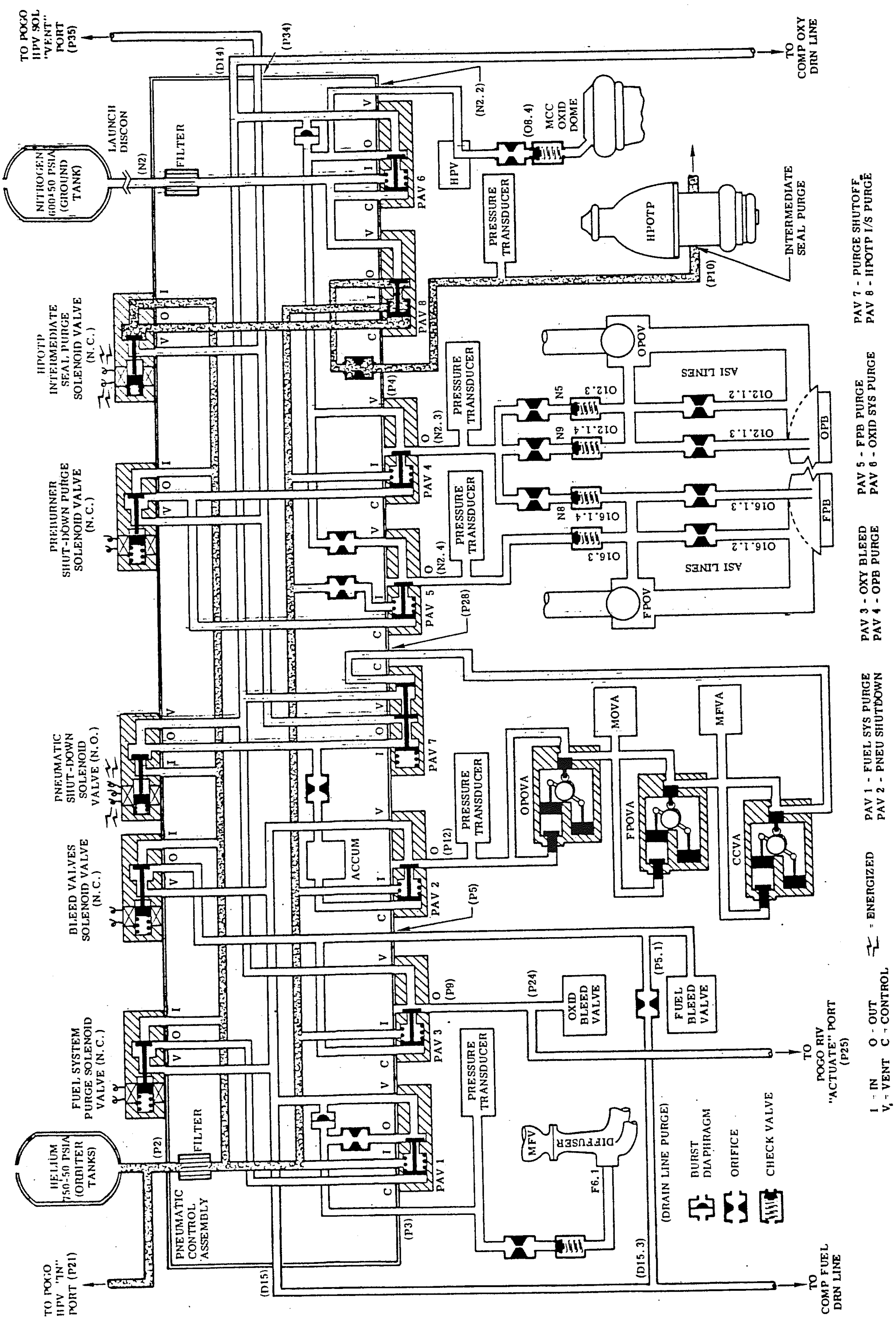
(c) Fuel system purge.

Figure 1.7-8.- Continued (3 of 12).



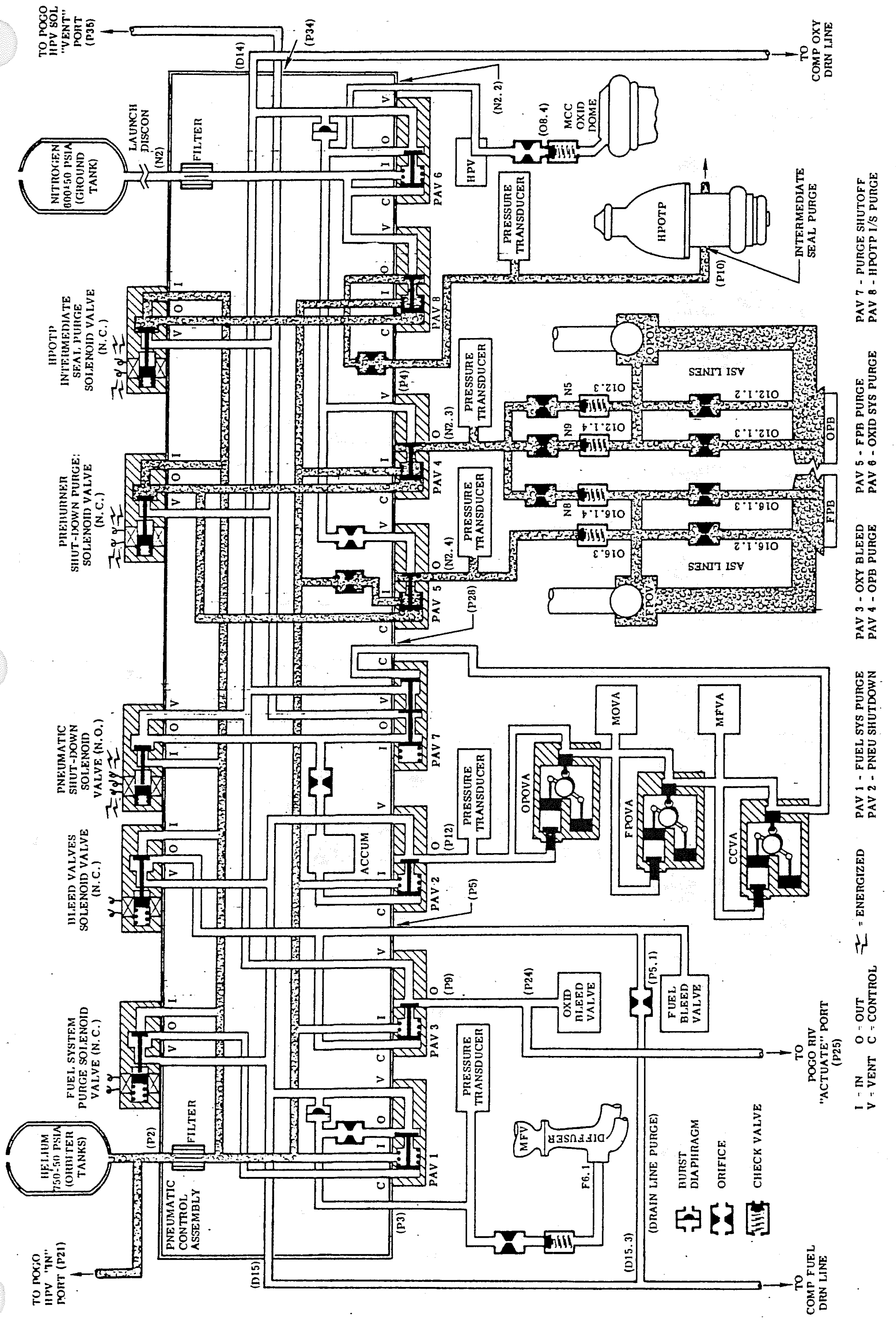
(d) Bleed valves open.

Figure 1.7-8.- Continued (4 of 12).



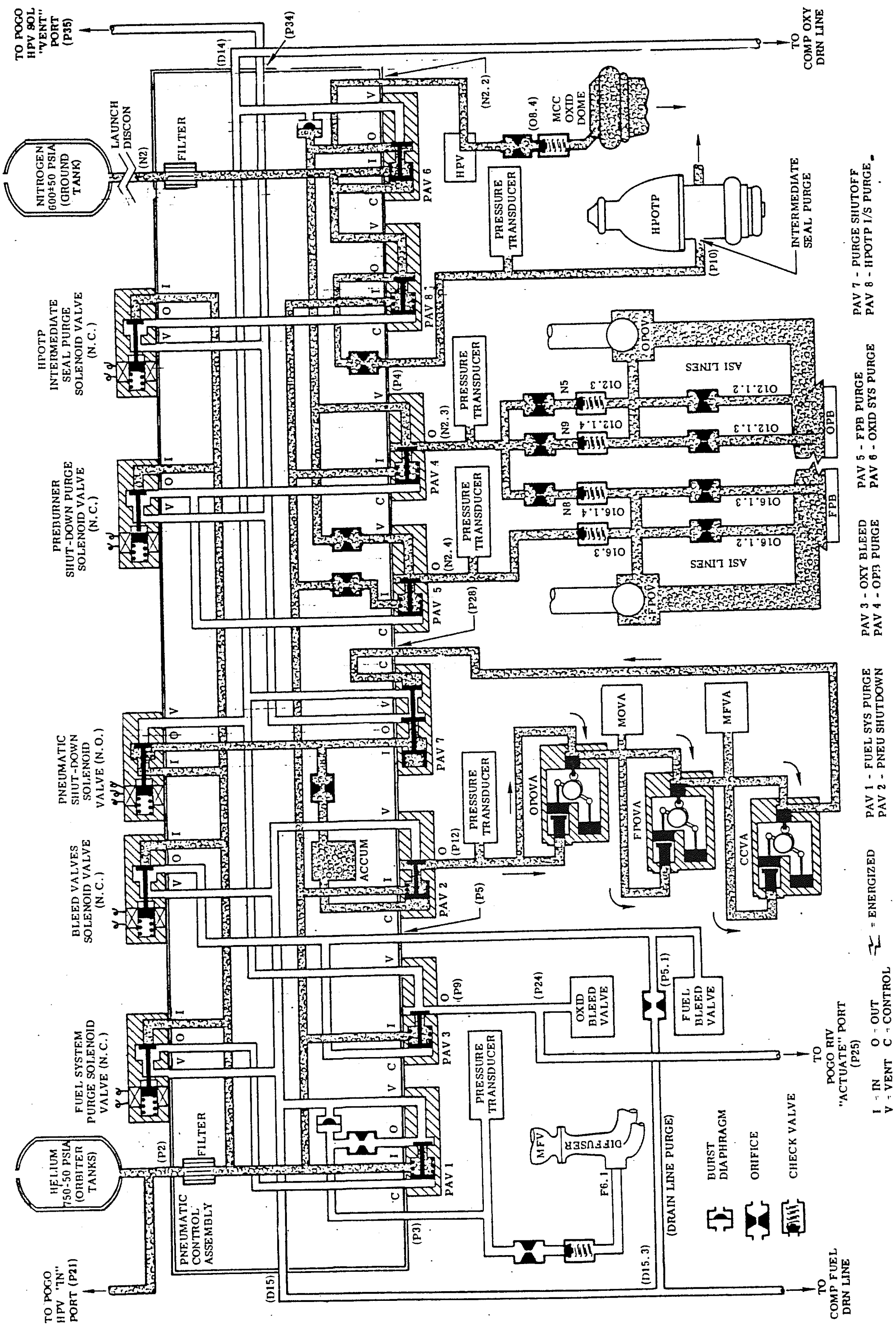
(e) HPOTP I/S purge (engine run configuration).

Figure 1.7-8.- Continued (5 of 12).



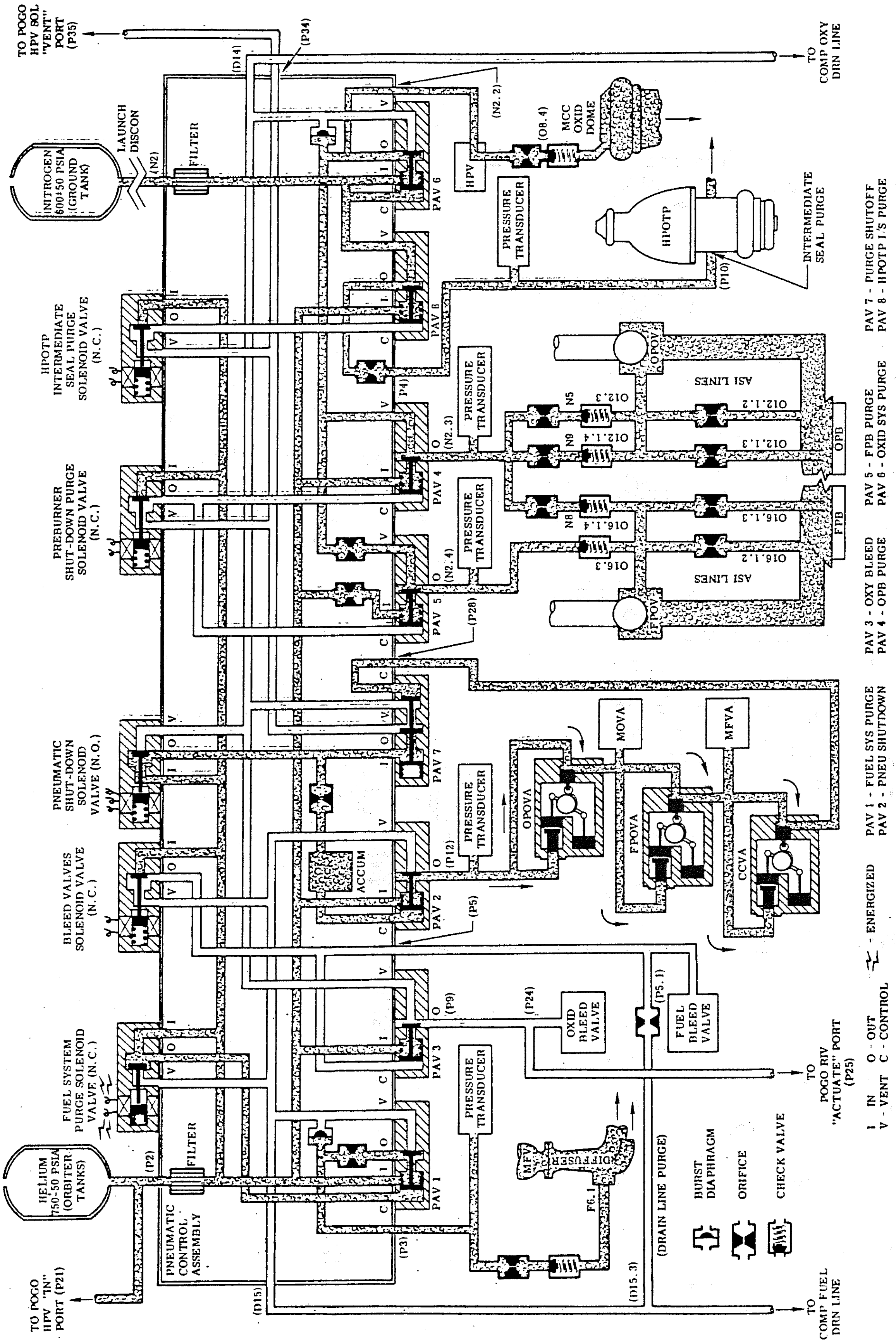
(f) Preburner purge (normal shutdown).

Figure 1.7-8.- Continued (6 of 12).



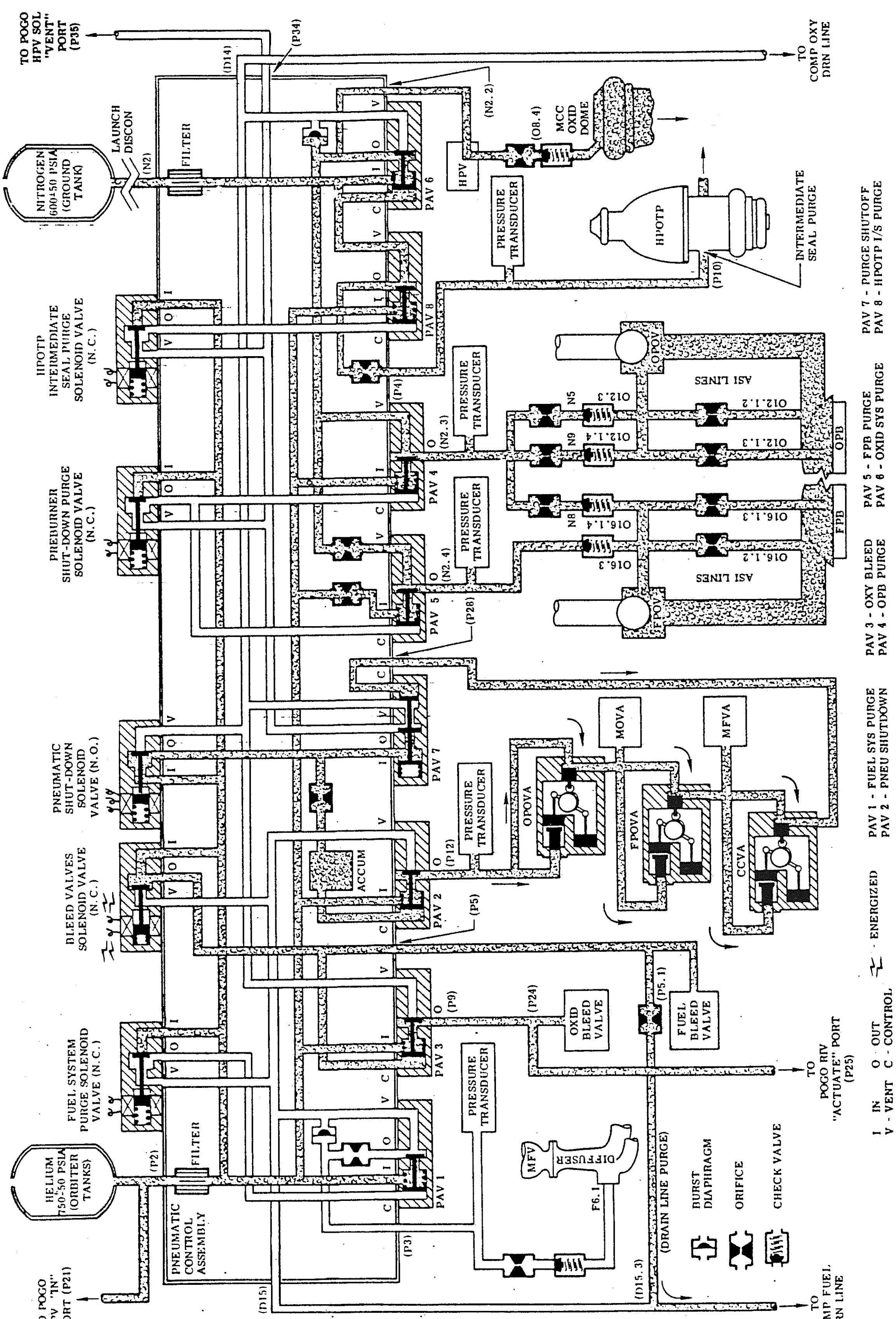
(g) Purge sequence no. 1 configuration.

Figure 1.7-8.- Continued (7 of 12).



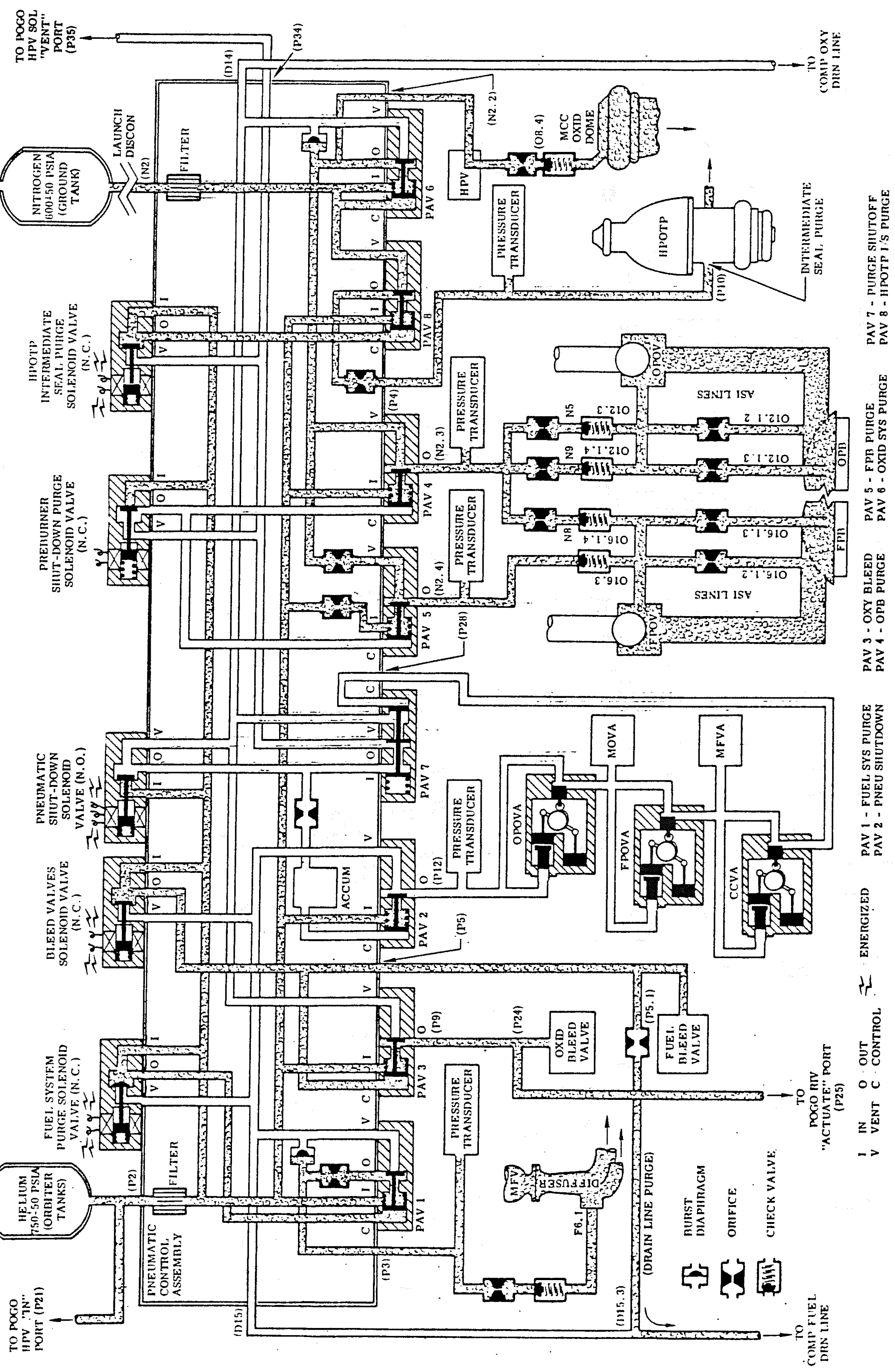
(h) Purge sequence no. 2 configuration.

Figure 1.7-8.- Continued (8 of 12).



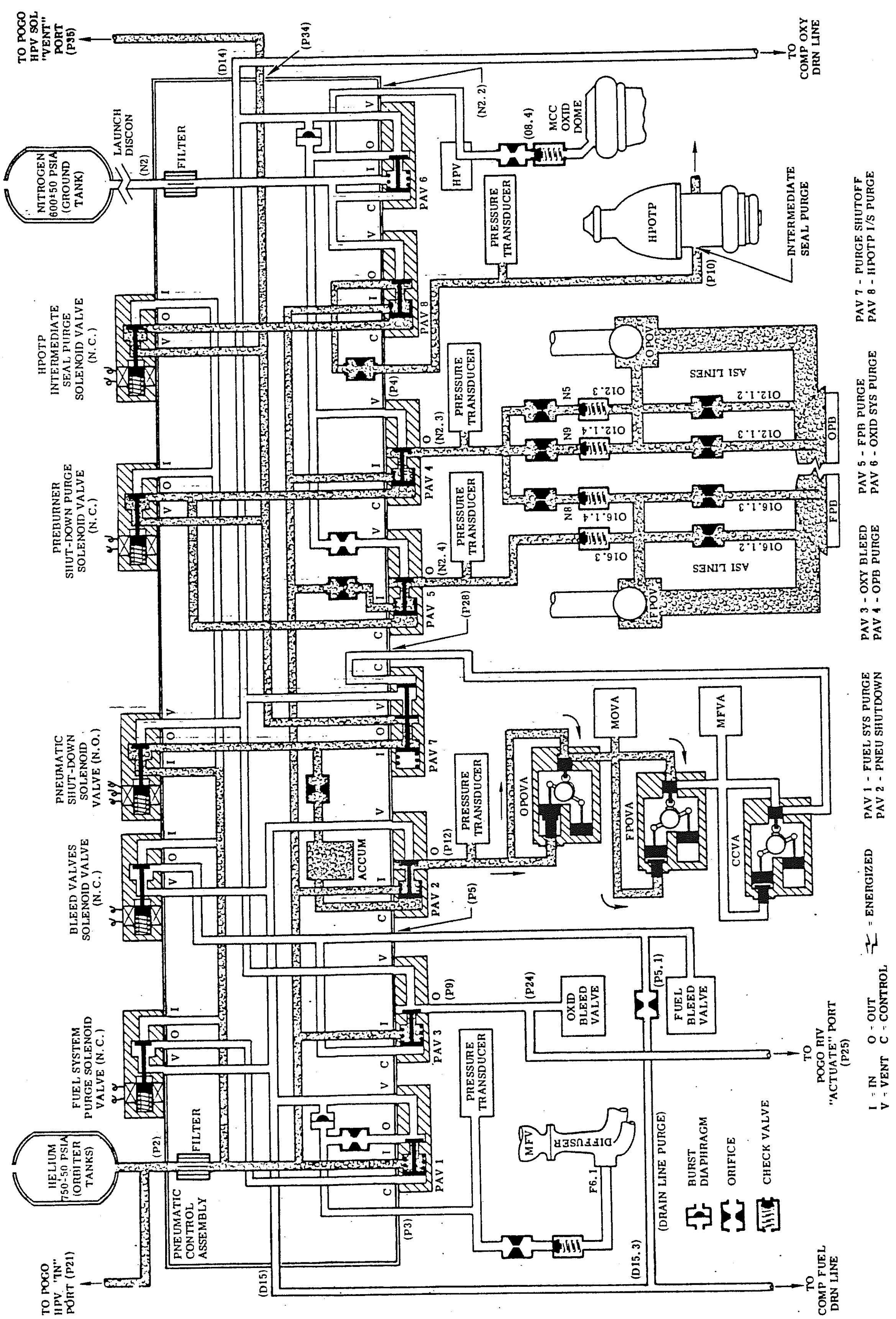
(i) Purge sequence no. 3 configuration.

Figure 1.7-8.- Continued (9 of 12).



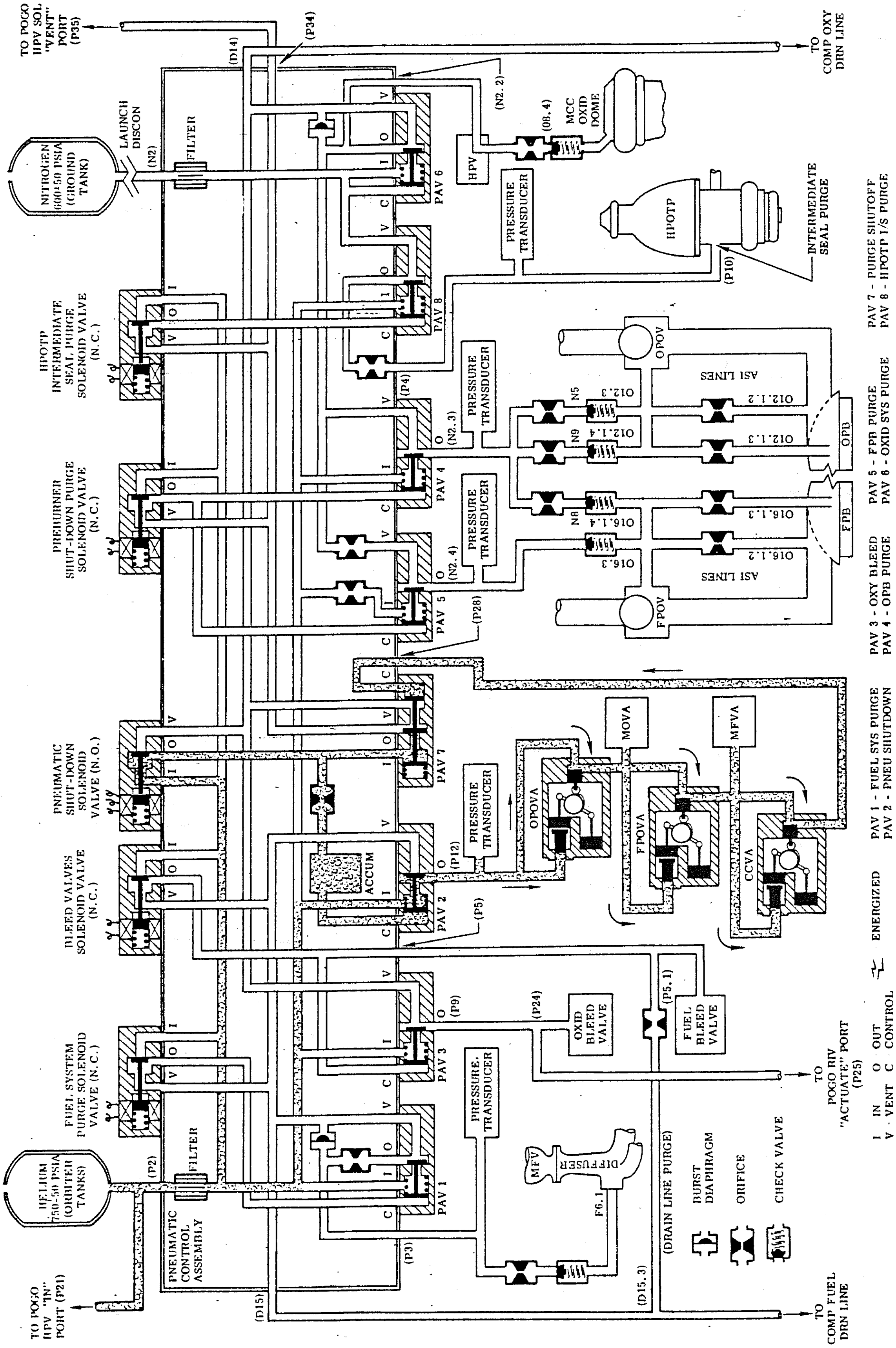
(j) Purge sequence no. 4 configuration.

Figure 1.7-8.- Continued (10 of 12).



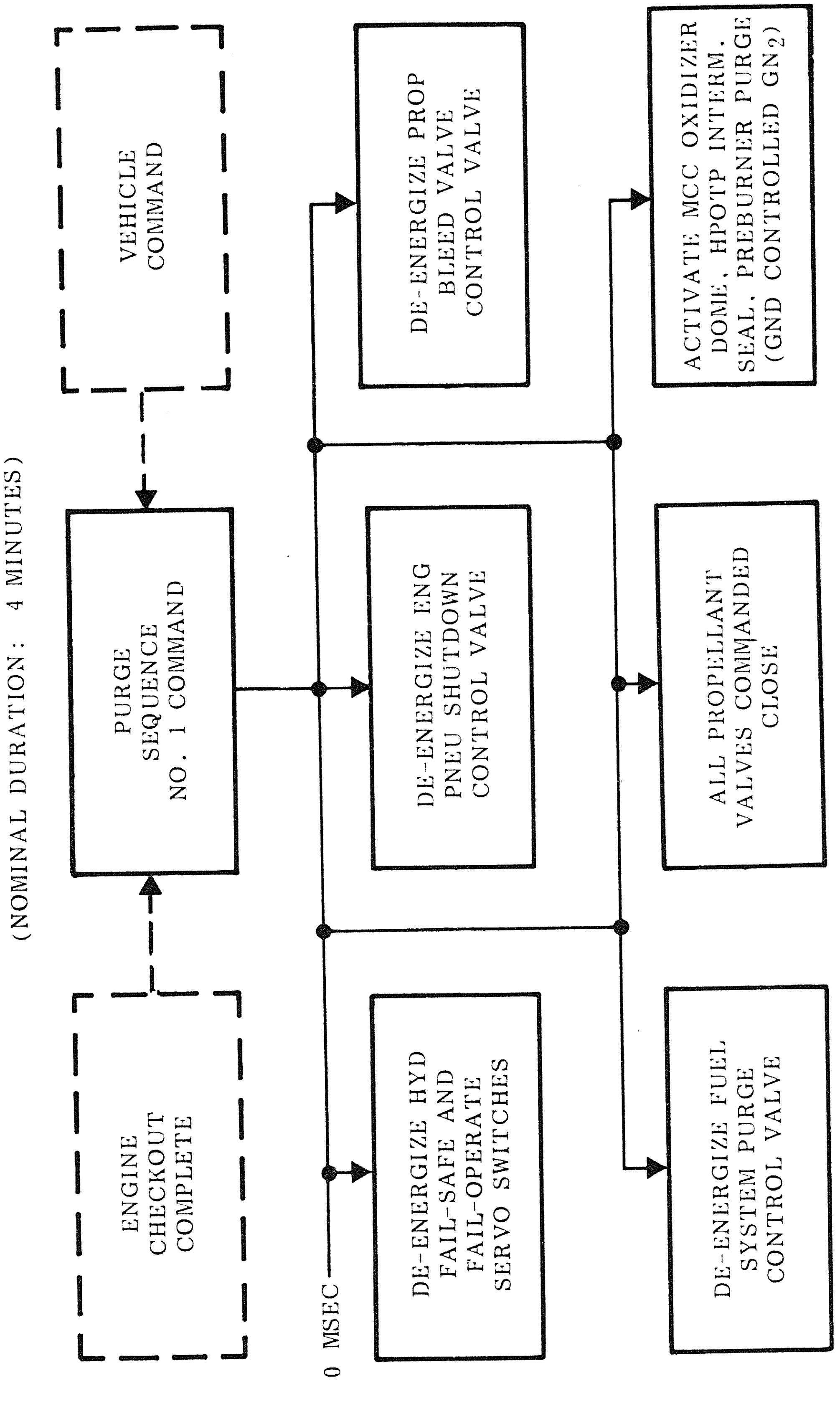
(k) Pneumatic shutdown; prop valves closing, purges on.

Figure 1.7-8.- Continued (11 of 12).



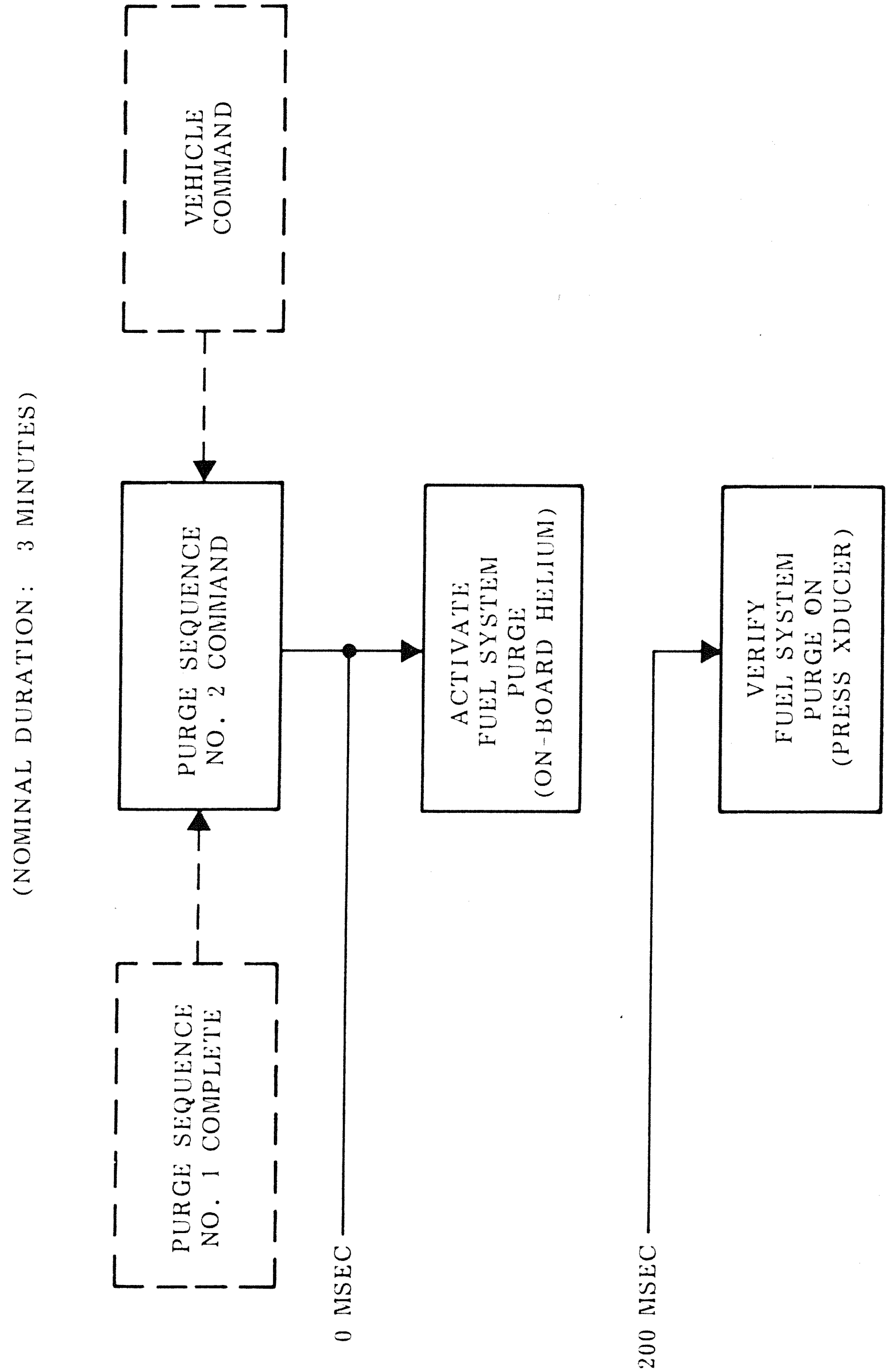
(1) Pneumatic shutdown; prop valves closed, purges off.

Figure 1.7-8.- Concluded (12 of 12).



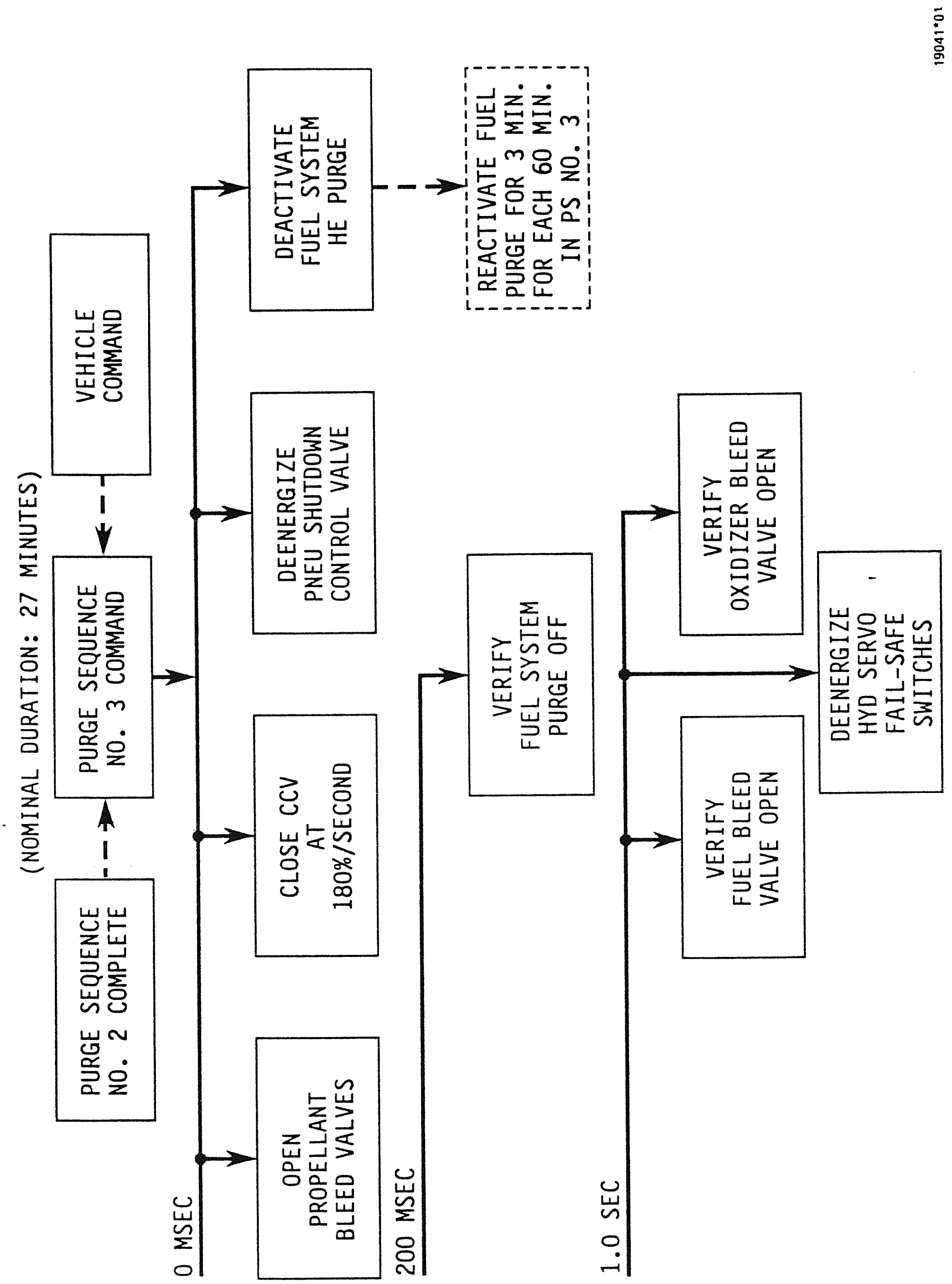
(a) purge sequence no. 1.

Figure 1.7-9.- Start preparation.



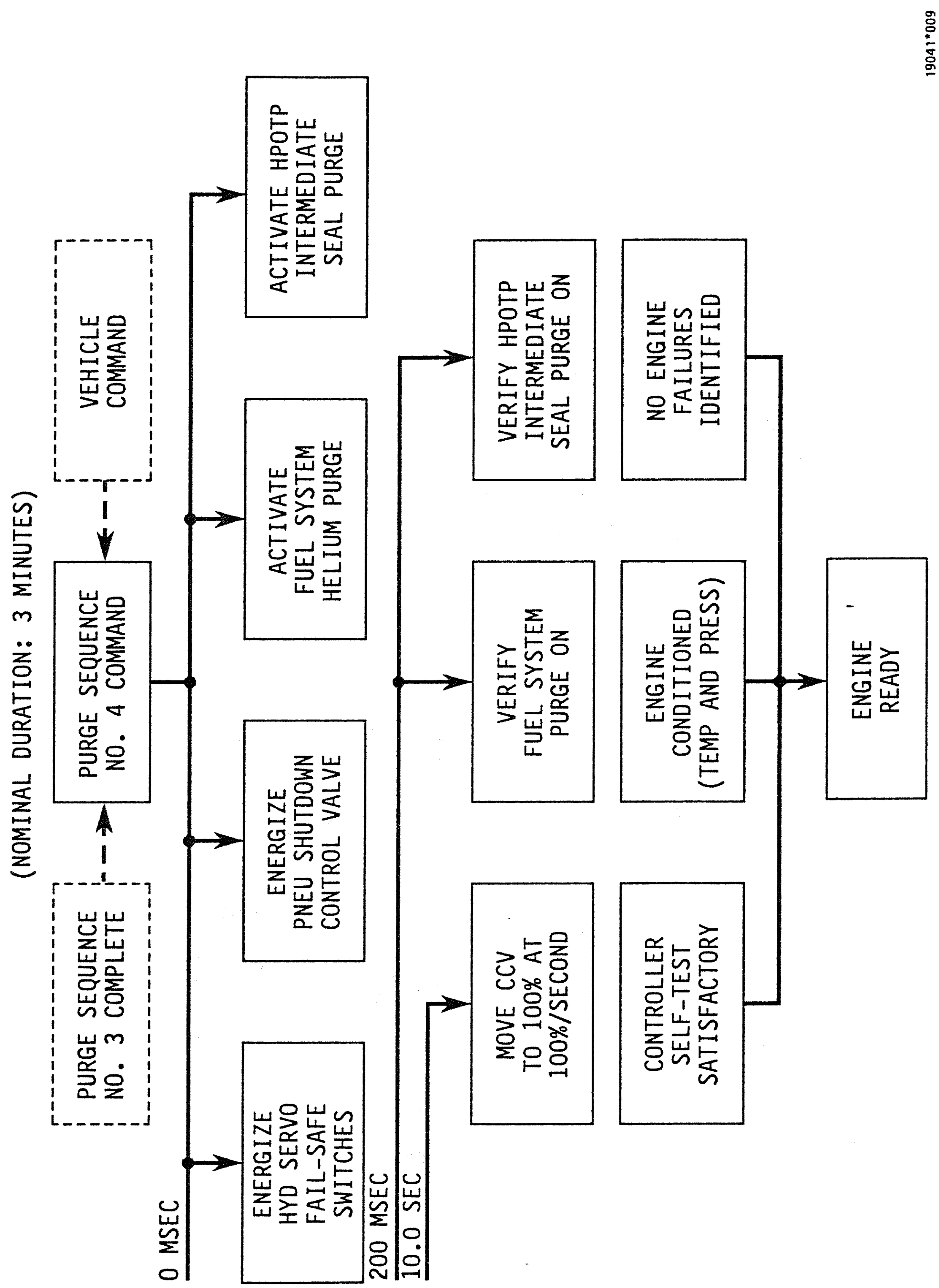
(b) Purge sequence no. 2.

Figure 1.7-9.- Continued.



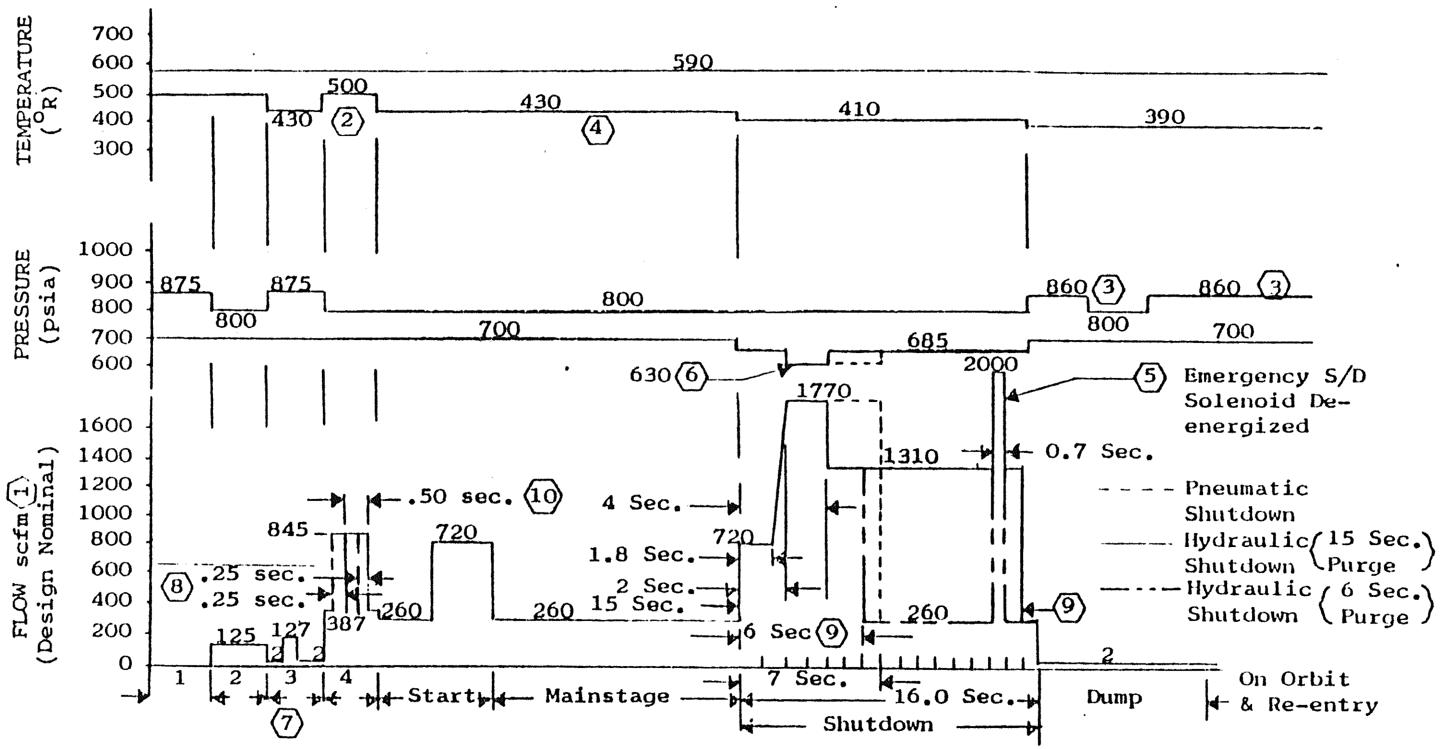
(c) Purge sequence no. 3.

Figure 1.7-9.- Continued.



(d) Purge sequence no. 4.

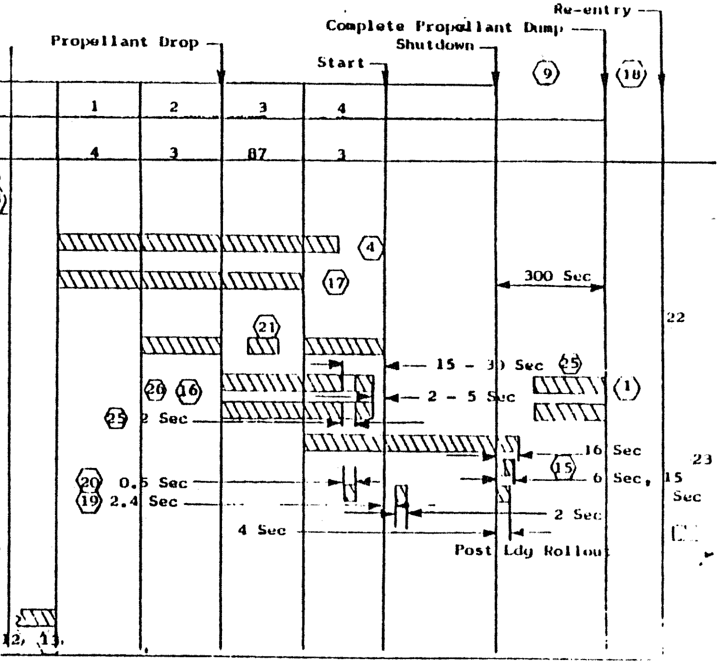
Figure 1.7-9.- Concluded.



NOTES:
1. Flow is Design Nominal and does not correlate with pressure temperature I/F variations.
2. Temperature requirements of 430°R maintained through the first 30 seconds of purge sequence No. 4.
3. For ground shutdown pressure values may reach 875 psia.
4. For RTLS abort the minimum Helium temperature is 395 °R and the minimum pressure is 685 psia from Engine Start +600 seconds through Engine Shutdown.
5. The I/F pressure and flow requirement are not applicable during this transient.
6. When pressure drops to 630 psia there is a corresponding decrease in flow.
7. Fuel System Purge on for 3 minutes each 60 minutes in Purge Sequence No. 3.
8. POGO precharge at Engine Start signal minus 15-30 seconds and 2-5 seconds (bleed valve cycle) prior to engine start signal. (STS-1 thru -4)
9. Six (6) second preburner purge shall be effective for STS-1 thru -15: Fifteen (15) second preburner purge shall be effective for STS-6 and Subs.
10. POGO precharge is on at "Start Enable" which occurs 2-5 seconds prior to engine start. (STS-5 and Subs)

Figure 1.7-10.- Engine pneumatic requirements (helium).

(3)	I/F No.	PRESSURE (psia)		TEMPERATURE (°R)		FLOW (scfm)
Purge Sequence No. (5)						
Minimum Purge Duration (Minutes)						
		Design (14) Nominal	I/F Variation	Design (14) Nominal	I/F Variation	Design (6) Nominal (14)
GROUND NITROGEN (4)(7) (24)						
Oxidizer Dome		600	+ 50 - 100	590 (7)	+ 10 - 20	265 ±20
HPOTP Intermediate Seal	(8)	600	+ 50 - 100	590 (7)	+ 10 - 20	65 ±
VEHICLE HELIUM (2)						
Fuel System Purge	(9)	750	See Fig. 5.1-3-1	530	See Fig. 5.1-3-1	125 ±12
Bleed Valve Actuator		750		530		—
Component Fuel Drain		750		530		2 ± 1
HPOTP Intermediate Seal	(9)	750		530		260 ±11
Preburner Shutdown Purge	(9)	750		530		1050 ±75
POGO Suppression	(9)	750		530		460 ±40
Oxidizer Inlet Feed System (11)	(1)	17.7	(13)	530	± 130	20 ± 5 (12)(13)
GROUND NITROGEN OR HELIUM (4)(7)(24)						
Oxidizer Inlet Feed System (10)	(8)	17.7	± 2	530	± 130	(He) 20 ± 5 (GN2) 7.5 ± 2



NOTES:

(1) Bleed Valves opened at post shutdown standby for duration of propellant dump sequence.
(2) Engine Helium consumption shall include additional helium per Figure 5.1.3-1 for each time the emergency shutdown solenoid is de-energized.
(3) The engine pneumatic requirements during FRT are identical to the operation requirements except for purge sequence duration requirements.
(4) The GN2 purge is ground controlled. GN2 purge activation will be simultaneous with purge sequence No. 1 command and will be within specified pressure limits in less than 5 seconds after receipt of this command. Deactivation of GN2 purge shall be such that the pressure requirements of Fig. 5.1-4 are met
(5) See Paragraph 5.1.
(6) Includes the effects of SSME hardware tolerances.
(7) Nitrogen purge heater operation must be "ON" prior to PSN-2 to insure the purge gas temperature is 590^{+10}_{-20} °R at the start of PSN-2. A minimum of 30 continous minutes of heated purge is required, of which 15 $^{+15}_{-0}$ minutes, shall be during that period after the start of LOX fast fill. and must be less than 550°R for a minimum of 30 minutes prior to engine start.
(8) Left blank intentionally.
(9) Post static firing and pad abort purge requirements are specified in Figure 5.1-8.
(10) Oxidizer inlet feed system purge is required on the ground at all times prior to purge sequence No. 1 except when liquid oxygen is present in the engine or when the HPOTP drains and thrust chamber exit (or throat) are covered, or when the system is required to be vented to accomplish engine checkouts or maintenance activity, or when the vehicle is in a controlled atmosphere per paragraph 5.1.
(11) Start of oxidizer inlet feed system purge is required prior to reaching an altitude of 100,000 ft. during re-entry. This purge is not required for an RTLS mission.
(12) Flow is applicable at an ambient pressure of 14.7 psia.

Figure 1.7-10.- Continued.

NOTES: (Cont'd)

(13) A purge flow of 4 SCFM Min. is required to ensure that the oxidizer feed system pressure is greater than local ambient pressure at all times. System pressure must not exceed a maximum of 50 psia.

(14) Design nominal parameters define the Engine flow control equivalent orifice area.

(15) Preburner shutdown purge is controlled by MEC and will not be applied during the period from start to start +2.0 seconds. For an RTLS the I/F conditions need to be met for 6.0 sec. only.

(16) Bleed Valves closed and initiate POGO Precharge purge at start enable signal at 15-30 and 2-5 seconds before start. (STS-1 thru -4)

(17) GN2 purge through the HPOTP intermediate seal is terminated by the SSME valves when the helium HPOTP intermediate seal purge is initiated.

(18) Post Flight purge requirements are specified in Figure 5.1-9.

(19) POGO Accumulator Post Charge is controlled by the MEC and will not be applied during the period from start, to start + 2.4 seconds.

(20) POGO Prestart purge is on at "Start Enable" which occurs 2-5 seconds prior to engine start.

(21) Fuel system purge on for 3 minutes each 60 minutes in purge sequence No. 3 (MEC controller).

(22) Appropiate measures shall be taken to assure that the SSME propellant valves are maintained closed prior to re-entry and during SSME postland repositioning.

(23) Six (6) seconds purge effective STS-1 thru -5. Fifteen (15) seconds purge effective at STS-6 and Subs.

(24) The allowable moisture content of the GN_2 source gas at the SSME interface is 6 ppm maximum.

(25) STS-1 thru -4)

(26) Bleed valves closed and initiate POGO precharge purge at "Start Enable" signal which occurs 2-5 seconds prior to engine start. (STS-5 and Subs)

Figure 1.7-10.- Concluded.

1.8 LO_2 PREVALVE CLOSURE AT SSME SHUTDOWN

1.8.1 General

When SSME shutdown occurs, the thrust, acceleration, and NPSP on the high-pressure LOX pump quickly falls to zero. In the valve shutdown sequence the main fuel valve remains open 5 seconds longer than the main oxidizer valve. This allows a fuel-rich flow to continue to drive the HPOT turbine while LOX flow to the high-pressure oxidizer turbopump is depleted. The absence of flow to the HPOT unloads the impeller causing the pump to overspeed. Since the LOX turbine and pump were designed to operate with a minimum head pressure, overspeeding due to the lack of head pressure will be catastrophic. To eliminate the possibility of this situation occurring, an artificial head is applied to the HPOT at SSME shutdown.

1.8.2 LO_2 Prevalve Timing

Timed closing of the LOX prevalves is critical for safe SSME shutdown. If the prevalves close too quickly the feedline manifold may overpressurize. If the valves close too slowly the HPOT may unload and overspeed causing a catastrophic shutdown.

Timing requirements were developed to protect against shutdown anomalies. These requirements considered worst-case dispersions of orbiter avionics and the SSME. The maximum allowable rotational speed for the HPOT is approximately 30,000 rpm. The burst speed is approximately 38,000 rpm. Based on the relationship between the time delay after SSME shutdown, HPOT speed and manifold pressurization, a target time delay was shown to be about 1.15 seconds. This corresponds to an HPOT speed of about 20,000 rpm.

After 51-L there was concern that a possible bit flip could give an erroneous phase indication, possibly leading to closing a prevalve on a running engine. This resulted in the establishment of the three-strike criteria. When the MECO status word has occurred and is read by the GPC, the GPC commands the MPS hardware to start the prevalve closing sequence. The LO_2 prevalves close 1.158 seconds after MECO during a nominal zero-g cut off and 4.5 seconds after MECO for pad aborts and premature cut offs. The GPC receives the MECO status word from the SSMEC via the EIU and reads it three times, a total of 120 ms, to make certain the command was legitimate. The GPC then sends the prevalve closure command to the orbiter MDM.

Note: An interruption in the data stream may cause the shutdown sequence to repeat. If the GPC has to read the MECO status word an additional three times, delayed prevalve closure may occur.

A data path failure produces the worse-case delay between shutdown and prevalve closure initiation. It takes two electronic hardware failures to cause a data path failure. If both failures or the second failure occurs after the second strike of the shutdown status word, the three-strike criteria cannot be met. This will delay the prevalve closure initiation for

280 ms. If both failures or the second failure occurs earlier and a data path failure is present in the same cycle the shutdown command is issued, the three-strike test will be bypassed and the prevalve closure sequence will be started 120 ms early.

1.8.3 Prevalve Function Redundancy

Electronically, the closing sequence of the LO_2 prevalves is single-fault tolerant on main bus failures and double-fault tolerant on flight-aft multiplexer-demultiplexers. Each prevalve has four FA MDM's and two main bus combinations in its circuit. This insures that only one prevalve will fail to close automatically when two main buses or three MDM's fail.

Mechanically the prevalves are single-fault tolerant. A pneumatic helium actuator is used to open and close the valves. The actuator is controlled by a series of four three-port valves. Each valve is controlled by a solenoid. Two of these actuator valves are set up in parallel on the closing circuit, only one is needed to close the prevalve. On the opening circuit there are two off valves in series, both are needed to open the prevalve.

Helium supply to the prevalve actuators is redundant. The primary helium source for the prevalves is the pneumatic tank. It can be interconnected to the left SSME He supply tank via the left engine He crossover valve. If necessary the center and right engine helium systems can be interconnected to feed the left system. As a backup, there are two 500 in^3 accumulators downstream of the regulators which has enough volume at 700 psi to close all LO_2 and LH_2 prevalves.

1.8.4 References

1. STS 83 - 0026A Sequencing Functional Subsystem Software Requirements

2. Space Shuttle System Handbook Drawing 10.8 - MPS HELIUM SYSTEM

3. Space Shuttle System Handbook Drawing 10.11 - MAIN ENGINE LO_2 SYSTEM

1.9 SSME FASCOS SYSTEM

1.9.1 General Description

The flight accelerometer safety cutoff system (FASCOS) was designed to provide vibration redline capability for the high-pressure turbopumps. The FASCOS redlines protect against bearing, impeller, and turbine blade failures when the system is connected and the redline monitoring is enabled in the SSME controller. FASCOS is currently deactivated for flights by disconnecting the FASCOS cables from the SSME controller, and patching the controller software to bypass FASCOS logic.

Each SSME has a FASCOS unit, which consists of six accelerometers, connecting cables, and the FASCOS box (fig. 1.9-1). Figure 1.9-2 shows the layout of the electrical components within the FASCOS box. Figure 1.9-3 shows how FASCOS fits into the overall SSME controller network. FASCOS is powered by three 28 V dc sources. Accelerometers A1, B1, and C1 are located at 0°, 174°, and 186° on the HPFTP pump. Accelerometers A2, B2, and C2 are located at 45°, 135°, and 135° on the HPOT preburner pump. The accelerometers are made of small pizoelectric crystals and weigh about 23 gm. The nonconducting crystals produce electrical output when subjected to mechanical stress. Figure 1.9-4 shows the locations of the accelerometers and their associated MSID's.

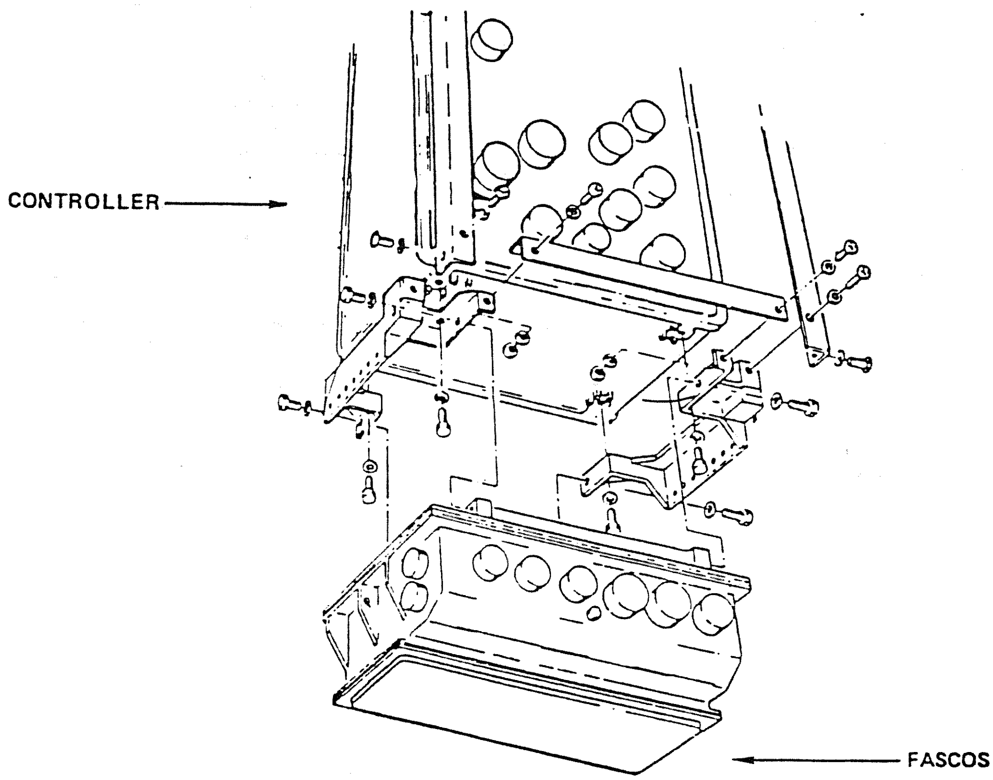


Figure 1.9-1.- SSME FASCOS.

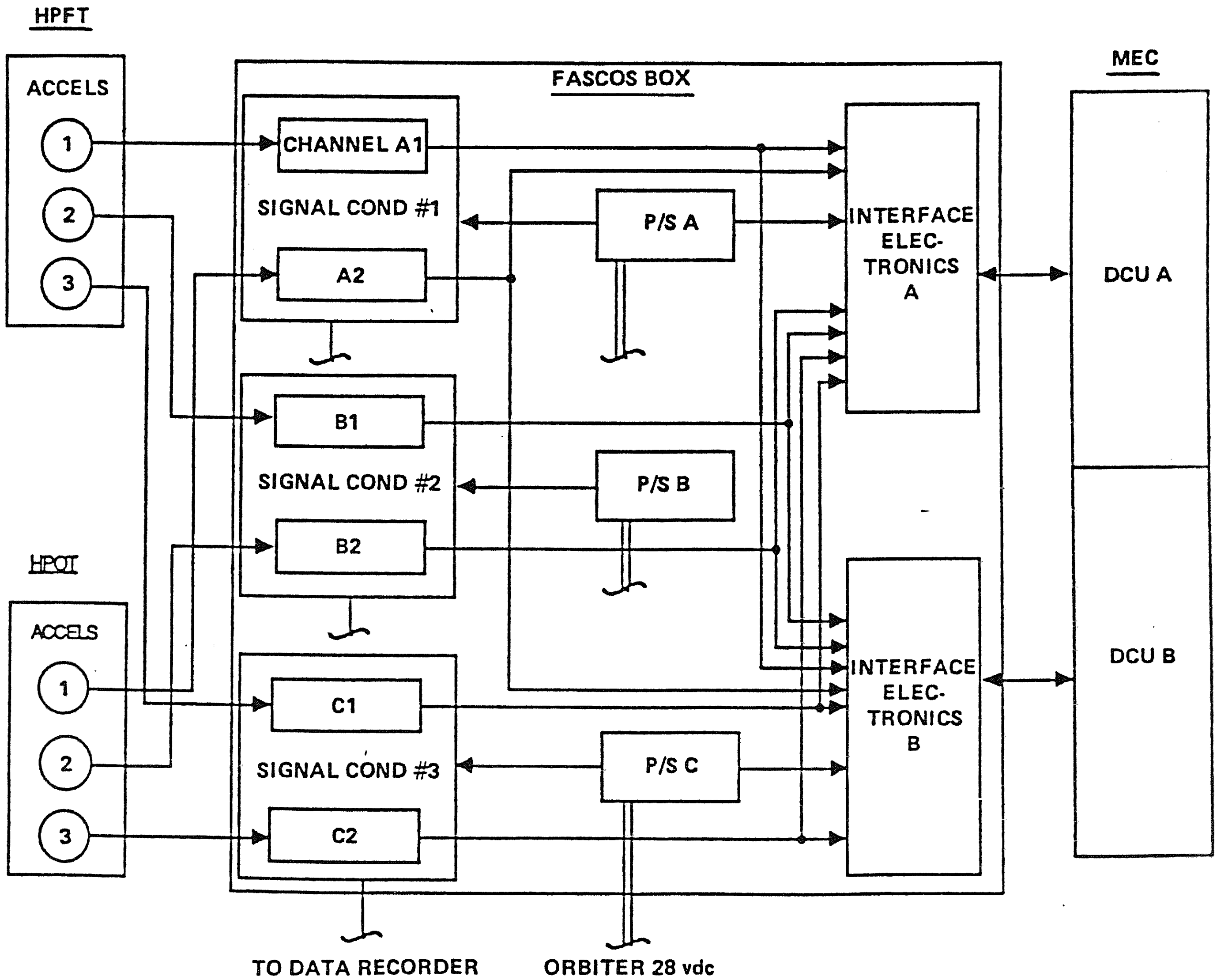


Figure 1.9-2.- FASCOS system block diagram.

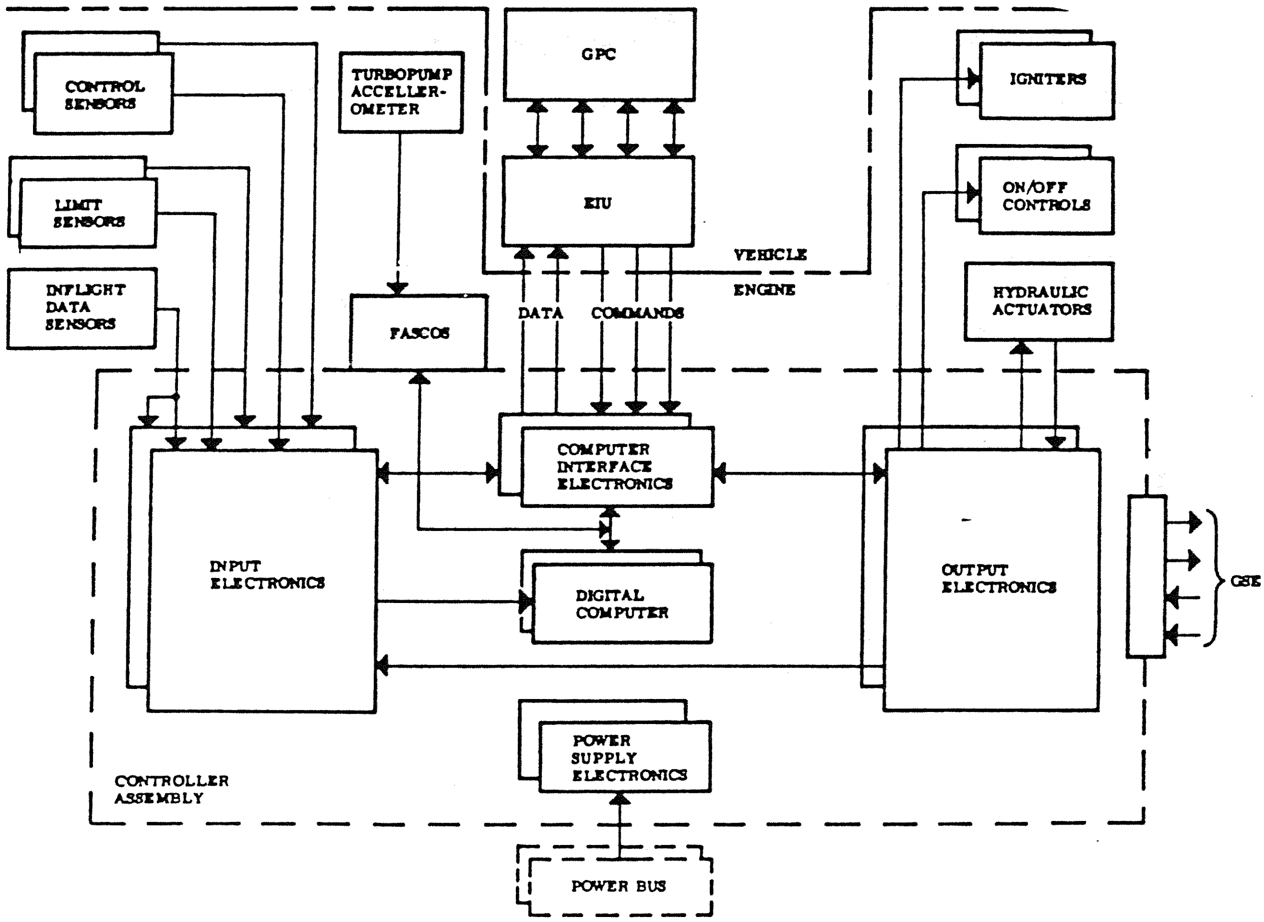
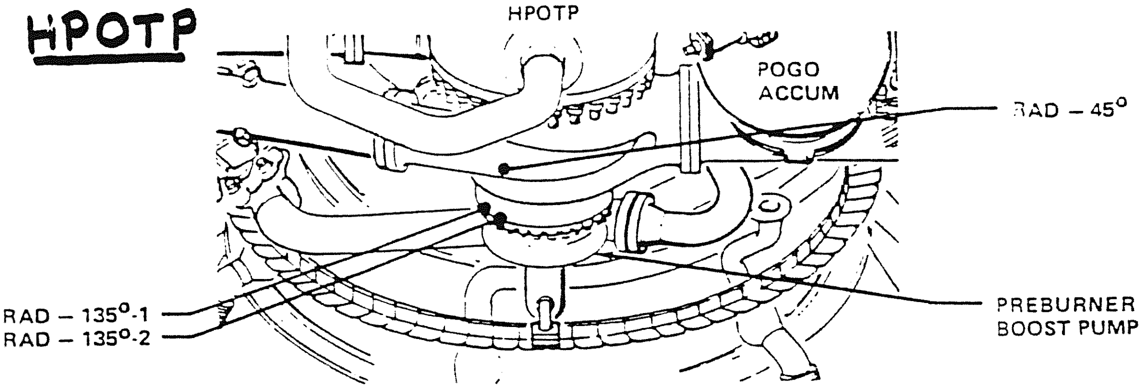


Figure 1.9-3.- Controller organization.

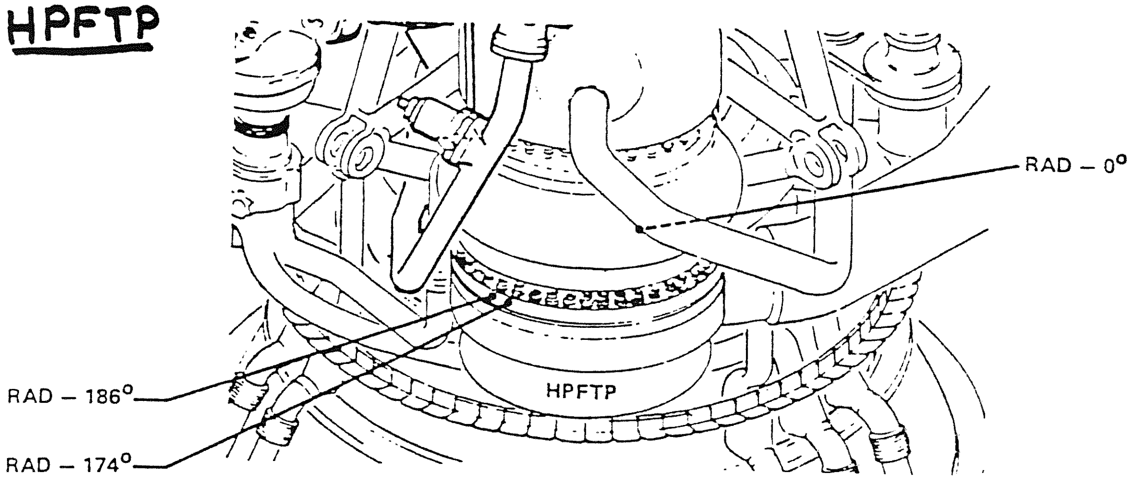
HPOTP



ASSOCIATED MSID(S)

E41DE601A FASCOS PBP RAD 45 CH A2 ON, -35 TO +35 G
E41DE602A FASCOS PBP RAD 135-1 CH B2 ON, -35 TO +35 G
E41DE603A FASCOS PBP RAD 135-2 CH C2 ON, -35 TO +35 G

HPFTP



ASSOCIATED MSID(S)

E41DE604A FASCOS HPFP RAD 0 CH A1 ON, -35 TO +35 G
E41DE605A FASCOS HPFP RAD 174 CH B1 ON, -35 TO +35 G
E41DE606A FASCOS HPFP RAD 186 CH C1 ON, -35 TO +35 G

Figure 1.9-4.- FASCOS radial accelerometers.

Raw accelerometer signals are sent to an electronics package referred to as the FASCOS box which is mounted with brackets on the SSME controller. In the FASCOS box, the signals are amplified to a working level and are passed through an 800 Hz low pass and a 50 Hz high pass filter (fig. 1.9-5). The signals, now in the 50-800 Hz range, enter an rms converter which calculates the equivalent g rms value. The g-level threshold detector then compares this rms value of pump vibration to a vibration limit. If the limit is violated, a "threshold exceeded" signal is sent to the SSME controller. Controller software determines if an engine should be shutdown by using the voting logic described in the next section. The SSME limit shutdown switch on panel C3 is used by the crew to enable/disable the FASCOS redlines in a way similar to the other redlines.

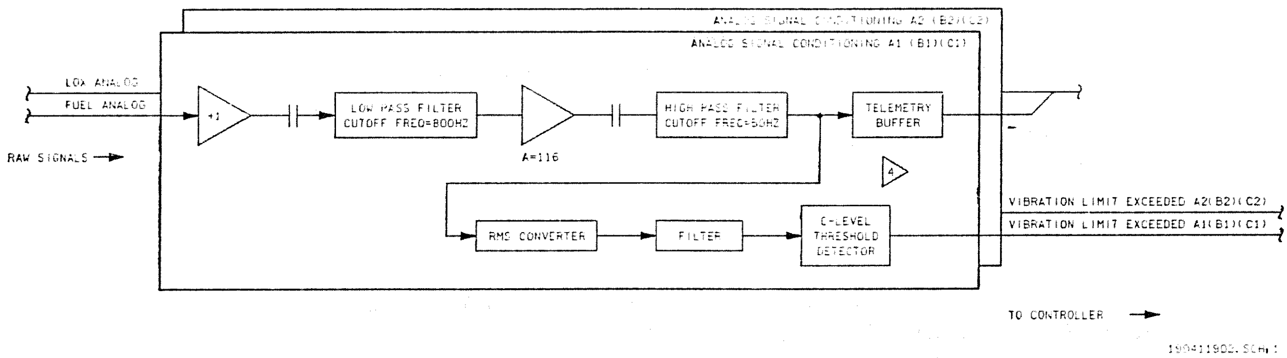


Figure 1.9-5.- Signal processing in the FASCOS box.
(from Space Shuttle Systems Handbook, 10.14)

1.9.2 Voting Logic - (Assumes logic in SSME controller S/W is active)

All voting logic is done in the SSME controller. Logic to initiate shutdown requires votes from three of three qualified accelerometers. If one channel is failed, two of two votes are required. If two or more channels are failed, FASCOS is disqualified, and the redline is terminated. To vote for shutdown, a qualified accelerometer must exceed the redline limit (16 grms for HPFTP and 11 grms for HPOTP) for three major cycles, (0.02 seconds/cycle). The voting accelerometers must exceed the limits simultaneously. A disqualified accelerometer is considered a vote for shutdown. If one of the channels fails on either the HPFTP or HPOTP, the corresponding accelerometers is automatically disqualified on the other turbo pump. (e.g., A1 pairs with A2, B1 pairs with B2, and C1 pairs with C2).

One vote for shutdown results in a FID 117 and the engine status word will indicate major component failure. Two votes for shutdown result in a FID 117 and the engine status word indicates major component failure. Also, FASCOS is disqualified, and the redline is deleted. If all three accelerometers vote for shutdown with limits inhibited, FID 117 appears, the engine status word indicates engine limit exceeded, and the engine does not shut down. If the limits are not inhibited, FID 17 will be posted, the engine status word will indicate engine limit exceeded, and the engine will shut down. If an accelerometer votes for shutdown, returns "in limits", and again votes for shutdown, no additional failure messages appear. Thus the ground controller is not notified of the actual vibration level, only the health of the FASCOS system.

1.9.3 Testing

The FASCOS controller undergoes a complete check when the SSME pneumatic systems are verified as early as 72 hours before launch. After these checks, there is limited failure detection by the SSME controllers. Before every data transfer, there is a "bit-toggle test" which detects 40 percent of FASCOS controller failures. Each channel is challenged to change a bit and change it back again. If one channel is dead, it is a default shutdown vote. If two channels are dead, FASCOS is disqualified. This test does not detect erroneous high or low vibration or engine-critical bus failures, but these failures would be evident anyway.

Qualification and shutdown monitoring processing for FASCOS is similar to the processing done for the other shutdown redline parameters. The crew inhibit/enable switch controls the shutdown logic for FASCOS as well as the remaining redlines. FASCOS processing, performed every major cycle by the DCU's, verifies the integrity of the hardware and monitors the status of the accelerometers. The FASCOS continuity/interface test verifies the functioning of the power supplies, accelerometer pairs, and addressing bits. This test is not done during commanded checkout tests or for the first 5 seconds following the start command; hence FASCOS is inhibited during this time. As long as FASCOS is qualified, the status of the bits is reported in the FASCOS status word bits 9 through 16 every major cycle. A failure must be present for three consecutive strikes to disqualify the associated hardware.

Failed elements are reported under FID 16 and are permanently disqualified until the next controller reset command. (A controller reset command is not sent during ascent.) The qualification processing uses the following criteria:

1. Accelerometers are disqualified in pairs.
   (A1 and A2, B1 and B2, C1 and C2)

2. Power supply failures disqualify the associated accelerometers.

3. Failure of bits 9 or 10 except by power supply failure disqualifies FASCOS.

4. Loss of two accelerometer pairs disqualifies FASCOS.

5. Loss of FASCOS/DCU A interface disqualifies FASCOS.

1.9.4 Conclusion

FASCOS is the only protection against machinery vibration. Analysis shows FASCOS prevents catastrophic failure modes and provides increased crew safety for any reasonable abort risk, but there is concern about using FASCOS as designed. While FASCOS could save a sick engine from catastrophic failure, it could also erroneously shutdown a healthy engine due to faulty electronics. The reaction time of the system is also in question; FASCOS might not kill the engine quickly enough to prevent failures. Using FASCOS requires a choice between the risk of losing an engine due to vibration and the risk of losing a good engine unnecessarily.

Engineering management was hesitant to use the newly designed system for STS-26. The young system, especially the sensors, had not yet demonstrated that it was fail-safe. There is a history of shutdown votes caused by faulty instrumentation. The previous 25 flights had 6 sensors which voted 1 of 3 to shutdown the engine. Therefore, in May 1988, it was decided not to use FASCOS for redline monitoring. The electrical connectors for the FASCOS redline threshold outputs are not connected to the SSME controllers. Only the filtered FASCOS 50-800 Hz accelerometer outputs to the orbiter MAD's are connected for monitoring turbopump health during flight.

1.10 SSME AUTO SHUTDOWN REDLINES AND REASONABLENESS TESTS

1.10.1 Start Confirm Redlines

There are certain parameters, known as start confirm redlines, that are monitored by the main engine controller to ensure a safe SSME start. If there is a redline violation during start phase, the engines will be shut down and a limit exceeded flag will be set in the RSLS. The redline values for each parameter are presented in table 1.10.I.

1.10.1.1 Main Combustion Chamber Pressure

The main combustion chamber (MCC) pressure (Pc) is a direct indication of SSME power level. The Pc limits are set to ensure that engine thrust is ramping up properly for launch at 100 percent (Pc = 3006 psi). Launching with an engine not delivering adequate thrust (low Pc) could require an abort to be performed. Engine thrust (Pc) is controlled by varying the amount the oxidizer preburner oxidizer valve (OPOV) is opened. The engine mixture ratio is maintained by the fuel preburner oxidizer valve (FPOV).

The overall average chamber pressure is used as the redline measurement. This is a single measurement required for launch. The redline values are as follows. From engine start plus 1.7 seconds to 1.74 seconds, the minimum value is 290 psia. From engine start plus 2.3 seconds to 2.34 seconds, the minimum and maximum values are 610 psia and 1000 psia, respectively.

Low MCC chamber pressures can be caused by an augmented spark igniter (ASI) failing to ignite. A low chamber pressure may also be caused by the failure of the OPOV to move or an erroneous RVDT signal to the controller which may prevent the OPOV from moving. Low chamber pressures may also be caused by an early priming of the FPB (causing low mixture ratios) which can prevent MCC and OPB ignitions. In these cases, the lower HPOT turbine discharge temperature redline will be violated. (Refer to paragraph 1.10.2.3 for this flight redline).

Partial HPOT cavitation will also be reflected in the chamber pressure redline exceedance. This can occur if the pogo relief isolation valve (RIV) fails to maintain proper gas/liquid level in the pogo accumulator.

1.10.1.2 High Pressure Fuel Turbopump Shaft Speed

This redline is implemented to ensure the high pressure fuel turbopump (HPFT) speed is adequate to provide the proper output for engine start and operation. This redline uses two sensors, both of which are required for launch (2 of 2 voting). The redline is implemented from engine start plus 1.24 seconds to 1.28 seconds and the redline value is a minimum of 4600 rpm.

TABLE 1.10-I START CONFIRM REDLINES

Parameter	Activation time (sec)	Limit	
		Min.	Max.
Main combustion chamber pressure (PSIA)	Start +1.7 to 1.74	290	--
	Start +2.3 to 2.34	610	1000
HPFTP shaft speed (rpm)	Start +1.24 to 1.28	4600	--
Antiflood valve position (%)	Start +2.3 to 2.34	80	--
Fuel preburner purge pressure (PSIA)	Start to start +2.28	0	50
Oxidizer preburner purge pressure (PSIA)	Start to start +2.28	0	50
Pogo precharge pressure (PSIA)	Start +4.94 to 4.98	800	1425

If the speed is reduced, the output is reduced, the amount of LH_2 supplied to the main injector will decrease, and the mixture ratio will be larger than required.

This can occur if there is leakage past the FPOV purge check valve. In this instance, oxidizer will be vented through the drain line causing a degradation of the ASI mixture ratio. The energy required to obtain adequate speed of the HPFT will not be generated resulting in a redline violation. A similar scenario can also occur for the FPOV ASI's failing to ignite (or outputting a low spark rate) or if the FPOV fails to move properly and is not detected by the controller.

A reduction of turbine power output can also be caused by leakage of the fuel preburner (FPB) seals. This results in reduced turbine speeds, flow and discharge pressure. This will also result in excessive HPFT turbine discharge temperatures.

The low pressure fuel turbopump (LPFT) which is upstream of the HPFT, has a definite effect on HPFT performance. An energy loss at the LPFT inlet will result in a reduced power output from the LPFT to the HPFT. The LPFT speed and output flow are decreased and the pressure delivered to the HPFT is reduced. The shaft speed will not increase sufficiently for safe engine start and operation. This will also result in excessive HPFT turbine discharge temperature.

1.10.1.3 Antiflood Valve Position

The antiflood valve (AFV), before engine start, is closed to prevent LO_2 from entering the heat exchanger. At engine start, the AFV opens to allow LO_2 flow into the heat exchanger (HEX). At this time, sufficient heat is available to gasify the liquid oxygen which is used to pressurize the LO_2 tank. If the AFV fails to open at engine start, there will be no flow into the HEX and no pressurization of the ET. Pressurization of the ET is required to maintain the structural integrity of the tank and to satisfy LO_2 NPSP requirements.

This is a single mandatory measurement which is monitored from engine start plus 2.3 seconds to 2.34 seconds. The minimum limit is set at 80 percent open.

1.10.1.4 FPB, OPB, and Pogo Pressures

There are three other parameters that are commonly referred to as start confirm redlines. These are the fuel preburner (FPB) purge pressure, the oxidizer preburner (OPB) purge pressure, and the pogo precharge pressure. These parameters are not considered start confirm redlines in the software but are required nonetheless for a safe engine start. A brief discussion of these measurements is presented in the following paragraphs. These parameters are also summarized in table 1.10-I.

FPB and OPB Purge Pressures - The chamber pressures of the fuel and oxidizer preburners is controlled by the fuel preburner oxidizer valve (FPOV) and the oxidizer preburner oxidizer valve (OPOV), respectively. (Refer to Booster Systems brief 1.2.) The FPB and the OPB are purged with gaseous helium during the start phase. Helium is routed from the pneumatic control assembly (PCA) through a check valve for each preburner. The check valve and the purge are present to prevent oxidizer from entering the PCA which would rupture the burst diaphram and vent into the oxidizer drain line resulting in contained engine damage. Also, if leakage is adequate to depress the augmented spark ignitor (ASI) mixture ratio, it is possible that the HPFT shaft speed redline will be violated. (Refer to paragraph 1.10.1.2.)

Pogo Precharge Pressure - The pogo precharge supplies helium pressure to the pogo accumulator during engine start until gaseous oxygen is available from the heat exchanger. Helium is routed from the pneumatic control assembly through the pogo helium precharge valve which controls the precharge pressure.

1.10.2 Flight Redlines

1.10.2.1 Main Combustion Chamber Pressure

The MCC Pc redline was developed in an attempt to protect against hardware failures that result in a reduction in Pc well below the commanded value. Abnormal degradation of main combustion chamber pressure may be caused by a drop in HPOT efficiency, reduced HPOT turbine inlet flow, loss of inducer/impeller head rise, or an energy loss in the inlet that reduces turbine power output. This results in reduced pump speed, flow, and discharge pressure which will cause Pc to degrade. The SSME controller uses a reference Pc value for each commanded power level. The controller will compare average main combustion chamber pressure to reference Pc and will sense the reduced pressure. The controller will increase propellant flow via the oxidizer preburner oxidizer valve (OPOV) in an attempt to increase chamber pressure. The OPOV will continue to open until Pc is in agreement with reference Pc. If the chamber pressure continues to drop, it is possible to reach an unstable power level (below 65 percent).

The redline value is reached when the average chamber pressure is 400 psi below reference Pc. This is done to prevent the engine from operating at an unstable power level.

Background

The failure that occurred during a test of engine 2106 in the oxidizer preburner (OPB) on July 1, 1987 prompted the addition of the MCC Pc redline. The failure was caused by a crack in an OPB interpropellant baffle pin braze joint. The effect was a hot streak in the OPB which eroded through the HPOT turbine housing causing approximately 20 percent of the hot gas to bypass the turbine. This bypass flow caused the overall engine power to be reduced. The test was terminated due to a ground test criteria violation. The damage to engine 2106 was contained, however, it is not known if the failure would be contained if the engine had continued to run. Therefore, an engine will be shutdown if both Pc pair averages have decreased 400 psi. This value was chosen because it equates to engine 2106 HPOT hot gas bypass flow.

1.10.2.2 High Pressure Fuel Turbopump Discharge Temperature

Excessive high pressure fuel turbopump (HPFT) turbine discharge temperatures may be caused by a drop in inlet pressure, distorted flow, or leakage past the preburner seals (static, rotor tip, platform, pump interstage seals) into the hot gas manifold. The reduced output or flow which can result is sensed by the main engine controller. The controller then increases FPB flow which will create excessive HPFT turbine temperatures. If not detected, this will cause uncontained engine damage.

An increase in head pressure is required at the inlet of the HPFT so that rotor imbalance and reduced turbopump output do not occur. Any type of imbalance, shaft movement, or vibration may result in rubbing of components. This will degrade the performance of the turbopump which will be sensed by the controller. Subsequently, the controller will increase FPB flow which will create excessive turbine discharge temperatures. Vibrations may also be caused by the failures of the turbine-end and pump-end bearings. These bearings are required to support the rotating assembly.

The LPFT which is upstream of the HPFT, has a definite effect on HPFT performance. An energy loss at the LPFT inlet will result in a reduced power output from the LPFT to the HPFT. The LPFT speed and output flow are decreased and the pressure delivered to the HPFT is reduced. The controller senses the increased demand by the HPFT and will in turn increase FPB flow resulting in high turbine temperatures as previously described.

Off-nominal density of the fluid (LH_2) will also be reflected in the turbine temperature. Proper fluid density is needed to prevent cavitation of the turbopump which will be reflected in increased temperatures. The fluid density will decrease if there is excessive leakage of GH2 past the turbine seal in the LPFT. The controller will increase FPB flow as before resulting in excessive turbine temperatures.

The upper redline limit is set at 1850-R and 1960-R for channels A and B, respectively. This redline was created as a result of two engine failures (0204,2013). A discussion of these failures is presented in the background. Subsequent analysis determined that the maximum turbine blade root temperature is 2160-R for 109 percent power level. This equates to a turbine exhaust temperature of 2060-R. The 1960-R limit provides a 100-R margin for the channel B location. The channel A location runs approximately 110-R cooler which results in the 1850-R redline. On some flights these redlines may be revised to ensure that there is a comparable margin between the predicted nominal operating temperature and the redline value.

Background

On October 7, 1981, engine 0204 failed during a test at 109 percent. The cutoff occurred after HPFT accelerometer redline exceedance. The probable cause was a thrown turbine blade or broken turbine nozzle which jammed the pump. This caused the pump to seize, resulting in a pressure surge and rupture of the low pressure fuel duct. The shutdown was lox-rich; the damage was uncontained.

On April 7, 1982, engine 2013 failed during a certification test at 109 percent. The probable cause of the failure was the loss of one or more turbine blades caused by the following scenario. A new bearing heat shield (coolie hat) retainer nut assembly was used for the first time on that test. The geometry of the nut caused a direct hot gas leak path through the bearing heat shield that impinged on the bearing coolant cap (kaiser hat). The bearing coolant cap then failed as a result of thermal stress and reduced structural properties and allowed hot gas to enter the bearing coolant circuit and starve the bearing coolant flow. As a result, the bearings' stiffness decreased causing increased synchronous vibrations. Synchronous vibration continued to build up until bearing failure followed by large rotor displacement, severe blade rubbing, blade loss, turbine seizing, fuel flow stoppage, and rupture of the pump inlet valve occurred. The resulting lox-rich shutdown caused a severe fire.

Prior to these two engine failures, a HPFT blade loss failure was thought to be a contained engine failure based on previous testing. The two engine failures prompted the development of a redline for the HPFT. This redline is based upon the high temperature blade life. Since the engine 2013 failure occurred right before STS-4 (June 27, 1982), a ground-observed redline was implemented for STS-4. Then on STS-5 and subs, the AUTO redline was implemented.

1.10.2.3 High Pressure Oxidizer Turbopump Discharge Temperature

A. Lower Limit

The lower limit of 720-R was designed to protect the engine from operating in an unstable region below 65 percent power level. This is due to the possibility of ice formation in the turbine which can result from 2 kHz clock failure in the SSME controller. If this clock fails, the output electronics (OE) would be disqualified. However, the chamber pressure monitor function would be erroneously affected. The chamber pressure would be stuck at the value present at the time of the failure and could never be disqualified. The controller controls thrust by controlling the chamber pressure with the oxidizer preburner oxidizer valve (OPOV). In this case, the engine would have problems when throttle commands are issued. After a command is issued to throttle down, the OPOV would start to close but chamber pressure would not change. As a result, the valves would continue to close in an attempt to bring the chamber pressure down to the proper value and ice formation will occur. Once ice forms, the pump could not function due to binding caused by the ice. The lower redline limit is designed to prevent this from occurring.

If the OPOV is prevented from opening properly, and if not detected by the controller, adequate oxygen for combustion will not be present causing the lower temperature redline to be violated.

Background

This redline was created after the engine 0010 failure where two failures occurred that drove the engine to operate at 1800 psia chamber pressure and a mixture ratio of 3.4. The two failures consisted of a controller channel failure and a Pc sense line failure. First of all, a controller 400 Hz power glitch caused channel B of the controller to be disqualified. Therefore, the Pc B pair ceased to be monitored. Secondly, the measurement port of the remaining channel A Pc pair became plugged by a failed purge control orifice (lee jet), and the Pc pair immediately stepped up in pressure to the purge supply pressure of 4300 psia. Since the controller controls the thrust by controlling the chamber pressure, which is about 3000 psia, the controller closed the OPOV to bring the erroneous Pc back to the reference value of 3000 psia. The resulting control point caused the engine to run at an actual chamber pressure of 1800 psia and a mixture ratio of 3.4 instead of the desired values of 3000 psia and 6.0, respectively. The HPOT balance cavity function at this exotic engine balance point becomes inactive allowing the pump shaft to be jammed towards the preburner pump end of the assemble. The power balance coincidentally allows the pump to run at or near its first critical shaft vibration resonance speed. This failure mode results in friction and/or bearing failure which leads to an HPOT detonation and an engine cutoff by the preburner pump accelerometer.

As a result of the engine failure, a hardware fix was made to the sense line, and the low limit of the redline was added. The hardware fix was a lee jet retention mod to prevent the lee jet from plugging the sense line if the lee jet breaks.

B. Upper Limit

The upper limit of 1760-R was designed to protect the heat exchanger from overheating and rupturing leading to uncontained engine failure. Engine analysis and testing indicate that the HPOT could run at 1860-R for one flight with an acceptable safety factor. Above 1960-R the heat exchanger could fail. The probability of failure is reduced at temperatures between 1860-R and 1960-R.

Excessive HPOT turbine discharge temperatures may be caused by erroneous MOV rotary variable differential transducer (RVDT) signals. There are two RVDT's for the OPOV (channel A and channel B). If the OPOV RVDT's are experiencing a slow drift which indicates the valve is full open, the controller will be mislead into taking action when none is needed. In this case, the valve is actually responding properly, however, the controller senses that the MOV is full open and compensates by closing the valve some finite amount. This will reduce oxidizer flow causing the lower redline to be violated. A similar argument can be made for RVDT drifts in the opposite direction causing high turbine temperatures.

A decrease or loss of fuel to the OPB will result in oxygen rich operation causing an increase in turbine temperature above the redline. Continued heating in this state will cause overheating of the turbine which can result in uncontained engine damage. This decrease in fuel flow may be caused by internal fuel tube leaks.

Turbine temperatures will also be affected by turbopump inlet conditions. A reduction in turbine inlet flow or an energy loss in the inlet reduces the turbine power output. This results in reduced pump speed, flow, and discharge pressure which will cause a degraded engine thrust or chamber pressure (Pc). The low Pc is sensed by the controller which increases OPB oxidizer flow which results in excessive turbine temperatures. This reduced inlet condition can be caused by leakage of seals within the HPOT (turbine blade tip seal, turbine interstage seal), OPB/HPOT seal, or excessive inlet flow distortion.

A loss of inducer/impeller head rise will also reduce engine thrust (Pc) as previously discussed. This is sensed by the controller which will increase the oxidizer flow to the OPB, resulting in excessive turbine temperatures as previously discussed.

The LPOT, which is located upstream of the HPOT, has a definite effect on HPOT performance. A loss of turbine power or head will reduce the output which is delivered to the inlet of the HPOT. The HPOT discharge pressure is reduced and MCC Pc decreases. This is detected by the controller which attempts to correct the situation by increasing oxidizer flow by opening the OPOV. This will cause excessive turbine temperatures as before, resulting in uncontained engine damage if not detected.

1.10.2.4 HPOT Intermediate Seal Purge (ISP) Pressure

Early in the development program, there were seal package failures that were uncontained. The seal packages were redesigned and redlines added.

The HPOT uses a dynamic seal design to separate the fuel rich hot gas in the turbine from the LO_2 in the pump. Mixing these two fluids results in imminent engine failure. The dynamic seal package uses helium as a working inert gas to maintain a positive pressure in the intermediate seal which creates a barrier between the LO_2 and the hot gas. This redline guards against excessive seal wear and/or loss of helium purge. The purge flow is routed from the orbiter helium supply bottles through the pneumatic control assembly (PCA) to the intermediate seal package. The low limit of 170 psia (measured in a remote spot on the pneumatics package) ensures that at least 110 psia is in the seal itself. If the seal fails due to a structural problem or the loss of the helium supply, uncontained engine damage can occur. For this reason, if redline limits are inhibited and the limit is being violated, immediate crew action is required to shut the engine down.

It is possible that complete engine failure will occur so quickly that neither the crew or the ground will have time to react.

1.10.2.5 HPOT Secondary Seal Pressure

The HPOT secondary seal pressure is a measure of the pressure at the HPOT housing of the combined leakage past the secondary seal and helium from the intermediate seal purge. This redline is implemented to ensure a positive helium flow through the intermediate seal package.

The secondary seal is part of the same dynamic seal package as the intermediate seal except it has no helium purge of its own. Instead it sees some helium pressure from the intermediate seal purge. The secondary seal high limit of 100 psia is designed to ensure that the minimum intermediate seal pressure is at least 10 psia higher (110 psia). This creates a positive helium flow to the secondary seal which is a drain cavity. Leakage past the primary or secondary turbine seal will allow hot gas to leak into the secondary seal cavity. This will increase the secondary seal pressure (and violate the redline) which will degrade the positive helium flow through the intermediate seal. The secondary seal redline can also be violated if there is a loss of coolant to the seals. The seals overheat and deteriorate resulting in a redline violation and uncontained engine damage if not detected.

A complete failure of this seal may not result in an engine loss with the limits inhibited since the hot gas must still pass the primary seal and the intermediate seal to get to the LO_2 in the pump. For this reason MSFC feels that, if the engine is running with limits inhibited and the secondary seal redline is violated, but the intermediate seal redline is not violated, the engine has a good chance of running until a safe abort region is reached.

1.10.2.6 HPFT Coolant Liner Pressure

The HPFT coolant liner pressure is a measure of the pressure between the turbine housing of the high pressure fuel pump and the coolant liner. Hydrogen is used to provide cooling to the main housing, turbine mount ring, turbine bearing support and turbine bearing support bellows. A rise in coolant liner flow will result in coolant liner pressure increase. This may be caused by excessive hot gas leakage into the hot gas manifold. A sufficient increase could cause the coolant liner to buckle, resulting in uncontained engine damage.

The coolant liner pressure is a known function of chamber pressure over the range of engine operation. Therefore, the redline is calculated real time in the SSME controller as a function of power level. The limits at 100, 104, and 109 percent power levels are approximately 3536, 3675, and 3850 psia, respectively. The redline is actually protecting the delta pressure between the coolant liner and the turbine exhaust. If this delta pressure exceeds 800 to 1000 psi, the coolant liner would buckle and collapse the exhaust gas turnaround duct. The HPFT would then stall and the fuel pump would stop, thus causing the engine to go LO_2 rich and sustain uncontained damage. The calculated pressure limit is designed to ensure that the delta pressure is less than 650 psia across the liner. Violating the limit would shut down the engine before coolant liner buckling could occur. The exact delta pressure at which a liner will buckle is unknown since it is a function of many variables including turbine exhaust gas temperature, material properties of each individual liner, and weld quality. For this reason, if limits are inhibited and the coolant liner pressure redline is violated, the ground would not advise the crew to shut down the engine until a safe abort region was reached. Since the engine is operating under these conditions, it is better to take the chance that it will continue to run until a safe abort region is reached. Attempting to predict when liner buckling will occur would be impossible in a real-time situation.

Background

The failure of development engine 0108 on February 14, 1984, prompted the addition of the dual channel coolant liner pressure redline on the HPFT. The engine was shut down by the HPFT turbine discharge temperature redline at 611 seconds into a scheduled 820-second 109 percent power level test. Before the engine could be safely shut down, the low pressure fuel duct ruptured and terminated fuel flow to the HPFT and the rest of the engine. The mixture ratio in the preburners and MCC became oxidizer rich and extensive damage was incurred in all combustion devices. The damage to the engine system was uncontained.

Preliminary failure analysis indicates that the most probable cause of the incident was a collapse of the HPFT turbine exhaust manifold with subsequent loss of power to the turbine, fuel flow stoppage and the low pressure duct rupture. Turbine exhaust manifold collapse was most likely caused by a turbine coolant liner failure because of high coolant liner pressure. Coolant liner pressure is metered into the liner cavity by 12 orifices in the turbine labyrinth shaft seal. If flow is diverted or bypassed upstream of the metering orifices, an unacceptably high pressure will result in the coolant liner cavity. It is conjectured that the bolts (12) securing the seal pack assembly, comprised of the lift-off seal, turbine labyrinth seal, and turbine aft platform seal, yielded allowing a direct bypass of fuel coolant into the liner cavity. Possible reasons for bolt yielding include increased temperature caused by extended operation at 109 percent power level or insufficient preload on the bolts for the increased 109 percent turbine speed.

1.10.3 Reasonableness Test

Each redline sensor is subjected to a reasonableness test. This is required to validate the sensor operation for application to the shutdown logic. The test is used to screen out a sensor(s) which may be outputting erroneous data (outside the normal operating range of the sensor).

1.10.3.1 Start Confirm Redlines

The reasonableness tests on all start confirm redlines reflect 3 sigma values of 25 flights and numerous ground tests. If the transducer is reading a value which is greater than this 3 sigma value it is voted out and engine will be shutdown.

1.10.3.2 Flight Redlines

1.10.3.2.1 Main combustion chamber pressure (Pc).- The reasonableness limit for channels A and B of this measurement is a difference of 75 psi from commanded Pc. This will put the engine in electrical lockup.

1.10.3.2.2 HPOT turbine discharge temperature.- The reasonableness limit of 2900 °R represents failed continuity of the transducer (open circuit due to a broken wire). The resistance goes to infinity and the transducer is no longer functioning properly.

1.10.3.2.3 HPOT ISP pressure.- The reasonableness of this measurement is a minimum of 0 psia and a maximum of 650 psia. This represents the operating range of the transducer.

1.10.3.2.4 HPOT secondary seal pressure.- The reasonableness of this measurement is a minimum of 4 psia and a maximum of 300 psia. This represents the operating range of the transducer.

1.10.3.2.5 HPFT coolant liner pressure.- The reasonableness limits for this measurement are a minimum of 1800 psia and a maximum of 4500 psia. This represents the operating range of the transducer.

1.10.3.2.6 HPFT turbine discharge temperature.- The upper reasonableness limit for this measurement is 2900 °R. The lower limit is based on a qualification limit calculation. This calculation is determined as follows:

```
Qual limit "calc" = ((slope) (PcRef) + offset) or 810 (use highest value)
           offset = (channel average) - (slope) (Pc) - 100
                    (calculated at 5.64 sec using 32 samples from 5.00 to
                    5.62 sec)
            slope = 0.1421 (A)
                  = 0.1809 (B)
```

The upper limit guard against open circuits when the connector wire breaks (infinite resistance). The lower limit protects against a degraded sensor.

1.11 SSME EIU HARDWARE

1.11.1 General

One engine interface unit (EIU) is located in each of the aft avionics bays (avionics bays 4, 5, and 6) in the orbiter. Each EIU acts as the information transfer interface between a dedicated SSME controller (SSMEC) and four GPC's. The EIU's weigh approximately 20.6 lb and have physical dimensions of 16.375 by 6 by 7 inches. Figure 1.11-1 is a perspective drawing of the EIU. Figure 1.11-2 is a functional block diagram of the EIU. A summary of EIU functions follows:

A. There is one EIU for each SSME. They are not interconnected, and they act completely independent of each other.

B. Each EIU acts as an information transfer interface between a dedicated SSMEC, located on the SSME, and four GPC's located in the forward avionics bays of the orbiter.

C. EIU's are used only during the prelaunch and launch phases of a flight. They are powered-up prior to SSMEC software loading and powered-off on-orbit (following OMS-2).

D. Commands (both execute and transfer) and command data words (associated with the transfer command) are received from four GPC's over four flight-critical data buses of the computer data bus network. See figure 1.11-3 for a diagram of the execute and transfer commands. Execute commands consist of a single command word and cause an EIU internal function to occur; e.g., "response" execute command causes SSME status information to be sent to the GPC. A transfer command consists of a command word followed by two command data words which are formatted into a SSMEC command word and sent to the SSMEC over three redundant SSMEC data buses. See figure 1.11-4 for GPC-to-EIU command and command data word formats. Figure 1.11-5 shows the format of the EIU-to-SSMEC command words.

E. SSME status data are transmitted back to the EIU from the SSMEC over two other redundant data buses. Upon GPC request by a "response" execute command, the EIU sends information back to the GPC's (only the first 32 words of the vehicle data table (VDT), figure 1.11-6) and automatically sends it (128 word VDT plus OIE overhead words, figure 1.11-7) through the EIU operational interface element (OIE) to the:

   1. Launch processing system (LPS) T-0 umbilical - Engineers at MSFC, KSC, JSC, and Rocketdyne/Canoga Park (RKD/CP) monitor the LPS data until the T-0 umbilical is pulled at liftoff.

   2. Maintenance and loop recorder number one - The data is recorded during ascent and the recorded data is dumped to Bermuda for review by MSFC and RKD/CP engineers.

3. S-band FM telemetry system - None of the OIE data are available in real time to JSC flight controllers. Only the limited data (32 words) transmitted to the GPC and downlisted in the GNC telemetry stream are available real time to the flight controllers. S-band FM is transmitted to Merritt Island and Bermuda during ascent and relayed by ground link to MSFC and RKD/CP for real-time monitoring.

F. Power is provided to each EIU from two of three redundant 28 V main dc power buses.

G. Cooling is provided by mounting each EIU on a coldplate.

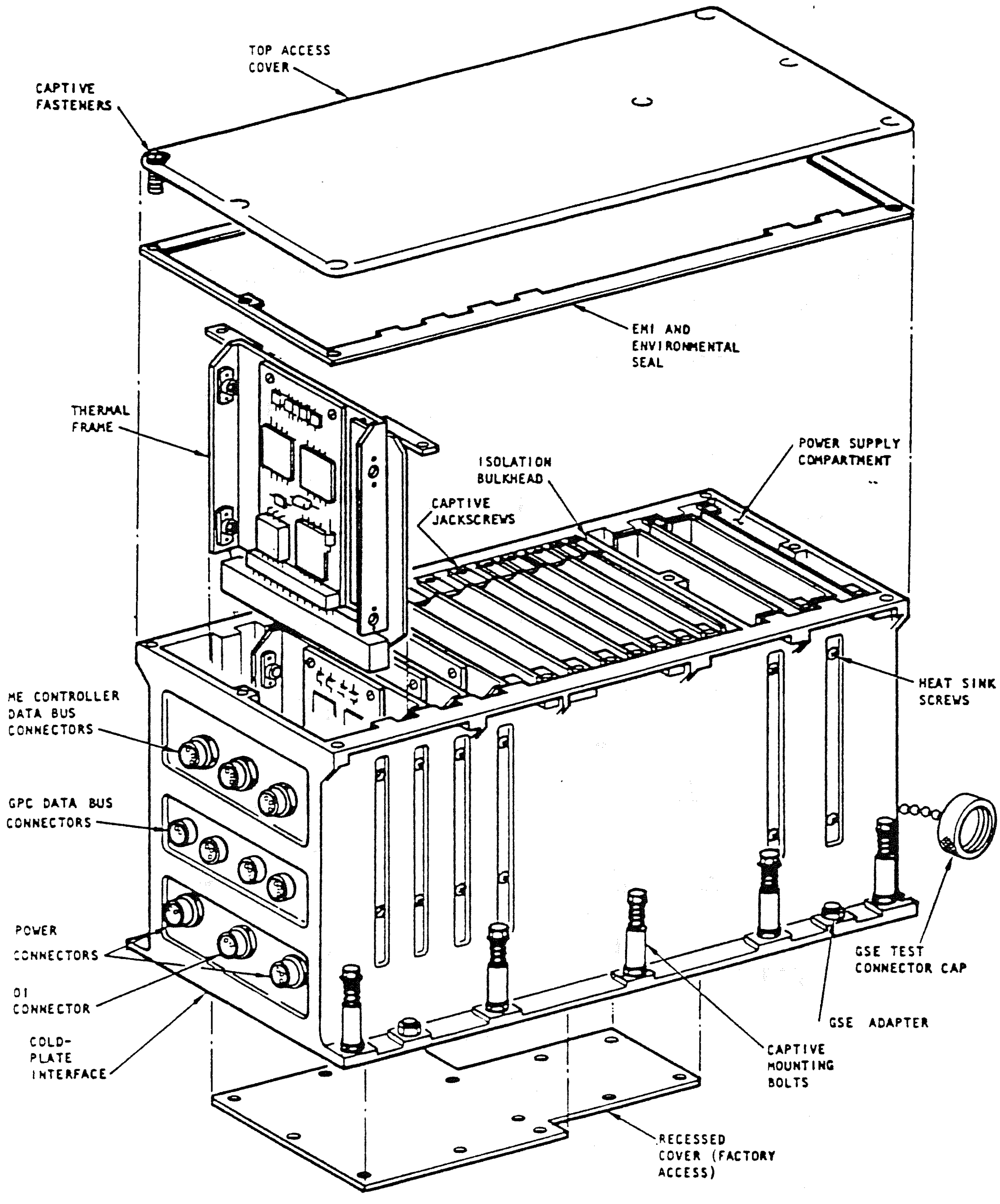


Figure 1.11-1. - EIU perspective drawing.

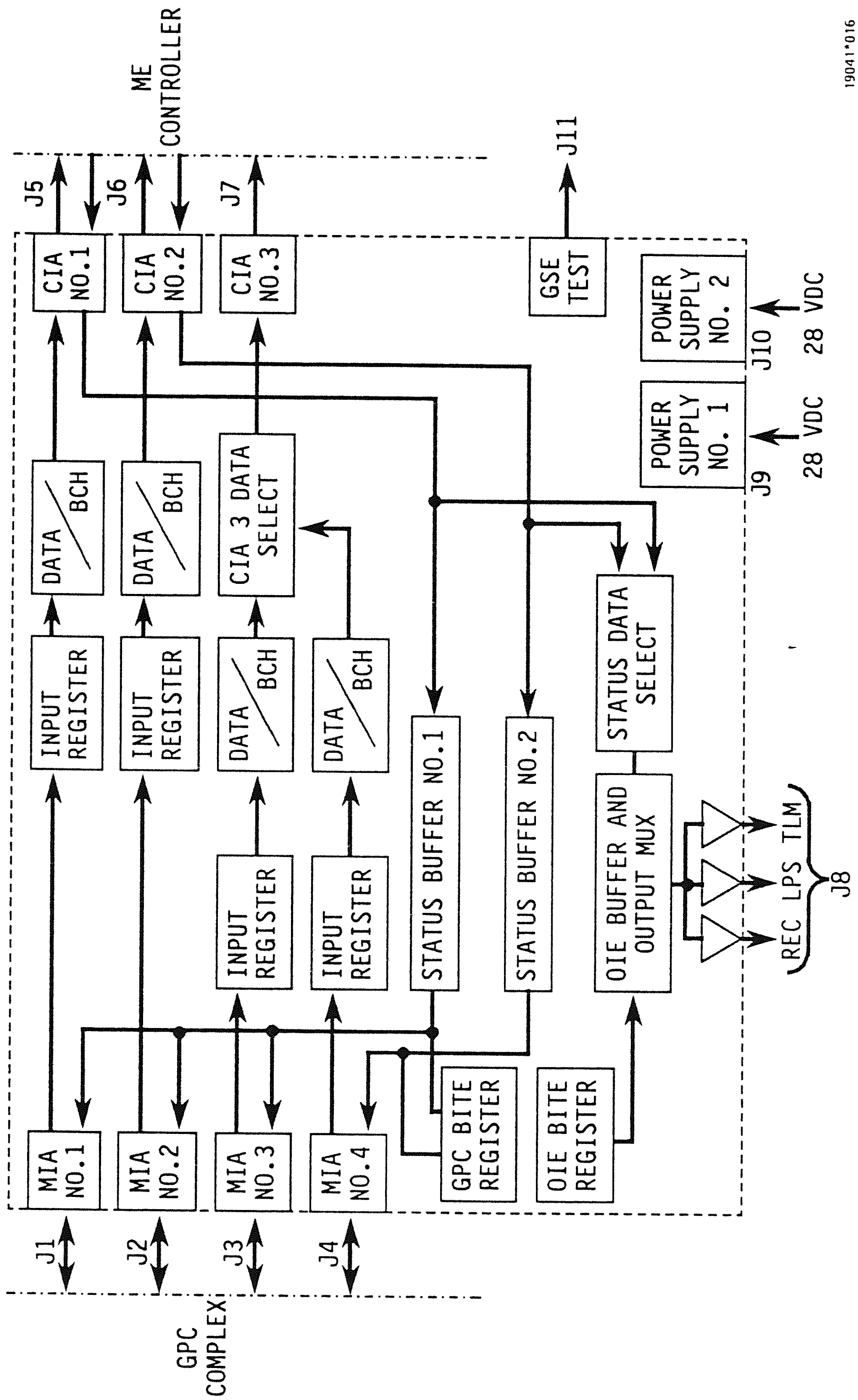
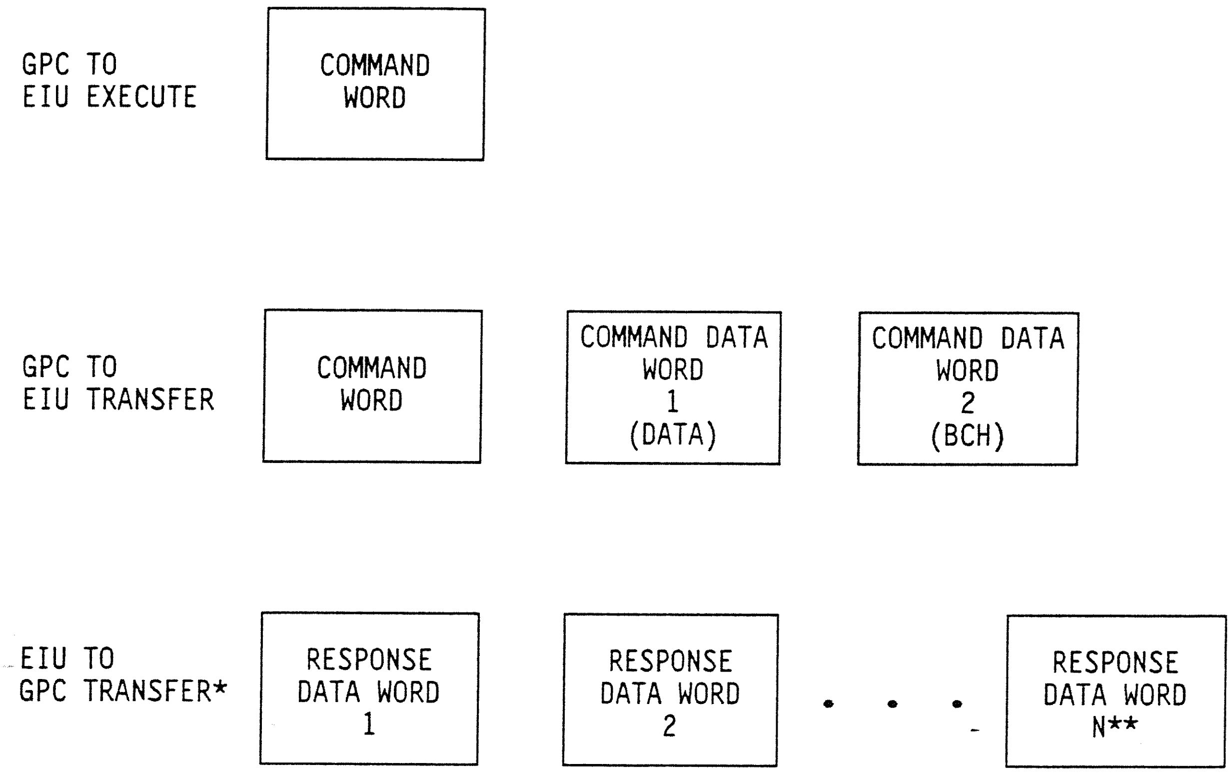
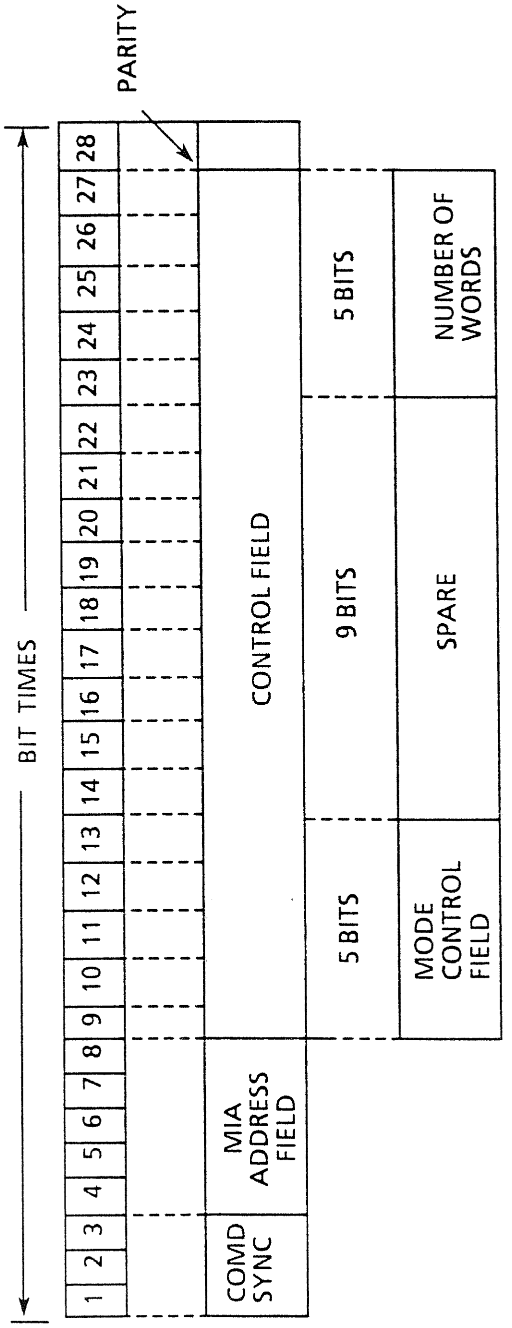


Figure 1.11-2- EIU block digram.

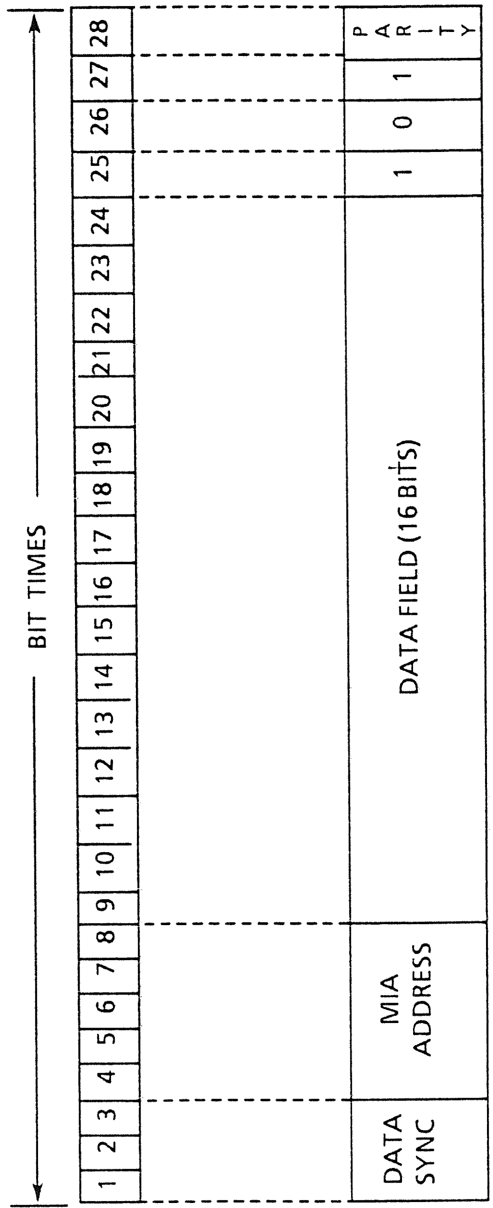


*IN RESPONSE TO AN EXECUTE COMMAND
**N EQUAL TO OR LESS THAN 32

Figure 1.11-3.- MIA message formats.



Command Word Format (form GPC to EIU)



Command Data Word Format (form GPC to EIU)

Figure 1.11-4.- Command and command data word formats.

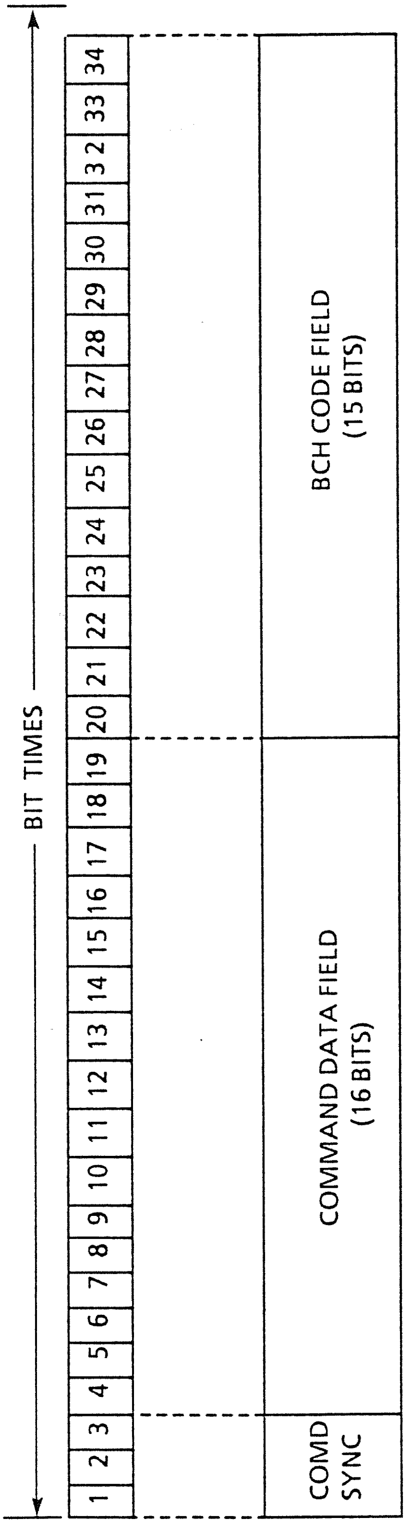


Figure 1.11-5.- Command word format (EIU to SSMEC).

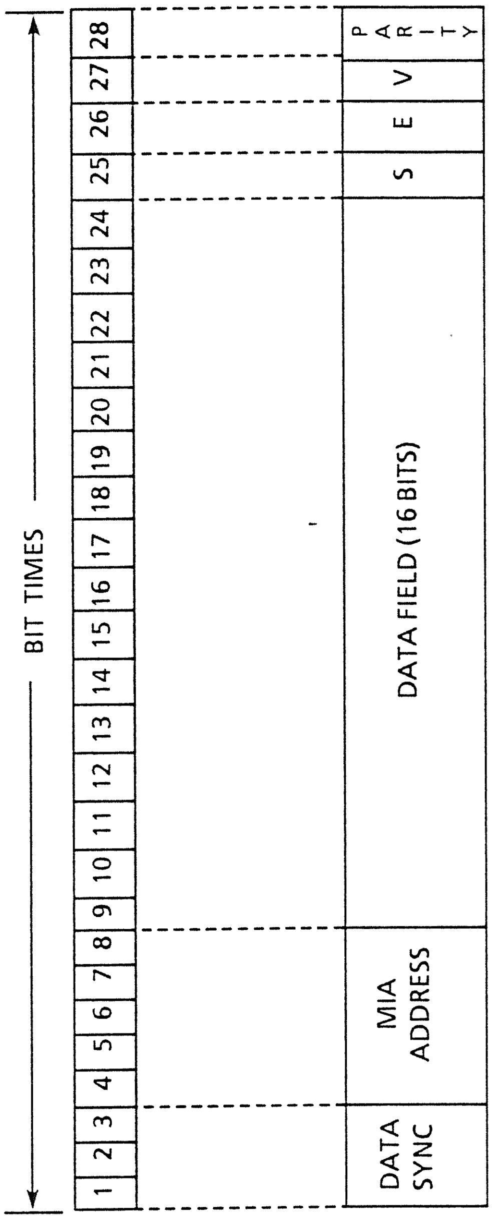


Figure 1.11-6.- Response data word format (EIU to GPC).

IDENTIFICATION	ME CONTROLLER DATA STREAM	BITE	COLUMN PARITY
2 WORDS	128 WORDS	1 WORD	1 WORD

Figure 1.11-7.- OIE message format.

1.11.2 Functional Description

The three EIU's operate entirely under the control of the GPC's, except that status or memory dump data is continuously received from the SSMEC, and automatically transmitted to the OIE. Each EIU accepts and decodes messages from up to four GPC's and transmits commands to its dedicated SSMEC when commanded by the GPC's. All signals entering and leaving the EIU are serial in nature and are encoded in the Manchester Bi-Phase format (non-synchronous). These non-synchronous signals have zero-to-one or one-to-zero transitions in the center of every data bit and this provides signal timing (fig. 1.11-8). Serial data transfer internal to the EIU is accomplished using the non return to zero (NRZ) format (synchronous). The EIU has an internal clock which it uses to synchronize NRZ transmissions (fig. 1.11-9). GPC control of the OIE data flow is limited to the selection of either the primary or secondary SSMEC status data channel.

The EIU is controlled by commands originated by the GPC and transmitted to the EIU over the digital data buses. The EIU is operational within 15 milliseconds after the application of power; however, messages containing words of all zeros are transmitted to the SSMEC until normal operations are initiated upon GPC command. OIE messages transmitted during the initial condition period will contain all zeros in the word count and data bits. The GPC must determine that an EIU is operational by transmitting a "return command" execute command and then checking the return message. The status of a unit is ascertained by the GPC through use of the "BITE" execute command.

1.11.3 EIU Major Components

The components of the EIU are shown in SSSH drawing 10.3. Each major component will be referenced to its location on that drawing and then its function will be briefly described.

A. Multiplexer interface adapters (MIA) 1-4 interface elements
   O-T Z 3-8

   1. MIA's 1-4   T Z 4-7

      The MIA's provide for bidirectional, serial, digital data flow between the EIU and the digital data bus network.

      Each of the four MIA's within an EIU has the same address, which is established by the wiring of the five address bits in each MIA's data bus cable connector. The addresses are created by selectively connecting the address pins to a logic one or a logic zero which is supplied by the EIU. The MIA's in EIU's 1, 2, and 3 have addresses 17, 23, and 24, respectively. The address status overhead bits are generated in the EIU for transfer and execute command/data word address checking and for response data word address encoding.

      Incoming data are transferred serially from the MIA to the MIA data register, and outgoing data are transferred serially to the MIA via the MIA interface element. Each MIA's receiver is always enabled, even while transmitting. This provides an immediate check on the transmitted word's MIA address validity.

      Each MIA is supplied with the SEV status overhead bits which it includes in each response word to the GPC. The S bit is the power transient flag and is set to a logic 1 for all transmissions. The E bit is the serial channel error flag which is set by the CIA logic for errors in response word. The V bit is the validity flag and is set to 1 for all EIU to GPC responses.

   2. MIA input register (1-4)   S Z 7

      The 24-bit MIA input registers provide the serial-to-parallel message buffering needed for reception of command words and command data words from the GPC via the MIA's (see figure 1.11-4 for command and command data word formats). Transfer and execute command words are broken apart and loaded into the MIA address compare, the command decoder, the timing control circuitry, 101 check pattern validity, and the message word counter. The first and second transfer command data words are loaded into the data transfer and BCH data registers in parallel for subsequent transmission to the CIA multiplexer (see figure 1.11-5 for the EIU-to-SSMEC command format). After receiving a "command return" execute command, the command word is serially transferred to the MIA interface element multiplexer for transmission to the GPC.

3. BCH and data transfer registers (data/BCH) P-Q Z 7

The EIU BCH/data transfer registers provide the message formatting and parallel-to-serial buffering needed for transmission of command and memory load words to the SSMEC.

The 16-bit data transfer register is loaded with the first command data word in a transfer command message. The 15-bit BCH register is loaded with the 15 most significant bits (MSB) from the second command data word. The 31-bit message is then serially transferred to the CIA for transmission to the SSMEC.

BCH is Bose, Chaudhuri, and Hocquenghem encoding which is used by the SSMEC's to determine if the command received is the same as the one transmitted by the EIU.

4. Word validation S-R Z 4-6

The word validation circuitry looks for errors in the words recovered by the MIA including invalid Manchester data encoding, parity errors, or invalid bit counts in the input words.

5. Command decoder and timing control R Z 4-6

The command decoder decodes the mode control bits which are transferred to it by the mode control command register in parallel. The time control provides the EIU internal logic with a 16-MHz clock signal which it uses to derive all other clock signals for synchronous data transfers between the EIU components. The time control also controls the timing (start and stop) of serial bidirectional data transfer between the MIA interface elements and the CIA's.

6. Interface element MUX (multiplexer) P Z 4-6

The interface element MUX takes the serial data from the status registers and builds the MIA response data words by adding the MIA address and SEV bits to the 16-bit words. It then serially transmits these response words to the MIA's so that the MIA's can add sync and parity bits and transmit them to the GPC's.

B. CIA 3 data select K-L Z 7-8

The CIA 3 data selection logic selects between the MIA 3 or MIA 4 transfer message (data/BCH represent SSMEC commands) to be transmitted over CIA 3 to the SSMEC. The logic selects the first channel to pass the MIA input validity checks for a three-word GPC-to-EIU transfer message. This function is also known as first-in/first-out buffering (FIFO). Once selection is achieved, it is maintained for 5 +/-1 milliseconds to ensure acceptance of the command data words following the command word. If both channels have data present at the same time, only one channel is allowed to transmit.

C. Built-in test equipment J-O Z 1-3

1. GPC BITE register J-K Z 1

The 16-bit GPC built-in test equipment (BITE) register provides storage for the GPC-oriented EIU BITE error signals. The register contents are transmitted to the GPC in response to the "BITE" execute command, and the majority of the register's bits are reset to zero. The "BITE" execute command is only exercised during prelaunch operations. The flags include MIA 2-MHz clock fail flags, CIA fail flags, status buffer fail flags, OIE buffer parity errors, power supply fail flags, format error flag, powerup sequence flag, and the status override flag. See table 1.11-I for a breakdown of the GPC BITE response word format.

The first GPC channel that requests BITE data obtains access and the other three channels are kept from processing the BITE command for a period of 50 microseconds by the inhibit BITE flags.

2. Power supply fault detector O Z 2-3

This logic determines if power supply A or B's +12 V, -12 V, or +5 V dc voltages are within nominal operating ranges. If an out-of-tolerance condition is sensed, the power supply failure flags are set in both the GPC and OIE BITE words.

3. Power-up detector O Z 1

The power-up detection logic is used to initialize the circuitry of an EIU after initial power up or after the loss and recovery of one of its power supplies during operation; i.e., just Power Supply A or B. The power-up detection logic is used to clear GPC and OIE BITE flags (except the power-up sequence bit in the GPC BITE register) and EIU output words (EIU-to-GPC, EIU-to-SSMEC, and EIU-to-OIE). When an EIU power supply makes a transition from a low voltage to a high voltage (>= 12 V dc), the power on reset signal associated with that power supply (A and/or B) is set to a logic 1 for approximately 100 microseconds. A power-on reset A will cause MIA port 1, MIA port 3, CIA port 1, Status Buffer 1, and the OIE to be initialized. A power on reset B will cause MIA port 2, MIA port 4, CIA port 2, and Status Buffer 2 to be initialized.

TABLE 1.11-I GPC BITE WORD BREAKDOWN

Bit times	GPC bite word
1	Synch bit 1
2	Synch bit 2
3	Synch bit 3
4	Most significant address bit
5	Most significant address bit
6	Most significant address bit
7	Most significant address bit
8	Least significant address bit
9	MIA 1 FAIL FLAG
10	MIA 2 FAIL FLAG
11	MIA 3 FAIL FLAG
12	MIA 4 FAIL FLAG
13	CIA 1 FAIL FLAG
14	CIA 2 FAIL FLAG
15	CIA 3 FAIL FLAG
16	Status buffer 1 FAIL FLAG
17	Status buffer 2 FAIL FLAG
18	OIE buffer FAIL FLAG
19	OIE control FAIL FLAG
20	Power supply A FAIL FLAG
21	Power supply B FAIL FLAG
22	Power up sequence FLAG
23	Format FLAG
24	Status override activated FLAG
25	Check field bit 1 (always 1)
26	Check field bit 2 (always 0)
27	Check field bit 3 (always 1)
28	Parity bit

Power-up detection logic was redesigned between the STS 51-L explosion and STS 26-R to eliminate a single point failure of the power-up detector circuitry which could cause a command/data path failure on the SSME being serviced by the EIU. A failure in the "OR" gate tied to the power on reset A and B signals could cause both of these signals to be tied to a 12 V dc source. This would assert these signals indefinitely and cause all of the MIA ports, CIA ports, OIE, and Status Buffers to be locked up in their initialization modes. All flights including STS 33-R and subsequent flights used the new configuration EIU's (dash 12 configuration).

D. Controller interface adapters (CIA's) 1-3   D-G Z 5-8

The CIA's provide for the simultaneous transmission and reception of data at a 1-MHz rate between the EIU and the SSMEC via the SSMEC data bus. In support of these primary functions, the CIA transmits dummy command words of all zeros when there are no commands to transmit to the SSMEC. It also performs verification checks for status words received from the SSMEC; e.g., Manchester validity checking.

Each of the three CIA's within the EIU is used to transmit commands to the SSMEC; only two (CIA 1, CIA 2) are used to receive data from the SSMEC. CIA 1 is the primary data channel and feeds data through status buffer 1 to MIA channels 1, 2, and 3. Although SSMEC data is available at MIA's 1, 2, and 3, only the GPC attached to MIA 1 reads primary data. CIA 2 is the secondary data channel and feeds data to MIA 4 through status buffer 2.

E. Power supply (A and B) and power source C   Q-S Z 1-3

The dual power supplies (A and B) in each EIU receive 28 V dc from two of the three primary dc power buses as shown in table 1.11-II.

Under normal operation, the dual power supplies share the EIU power requirements through three circuits, A, B, and C as shown in table 1.11-II. A and B are two single supply circuits, and C is an OR'ed (A or B) supply source. The three supply circuits are distributed within the EIU so that a failure of one circuit will not prevent the EIU from processing communications between the GPC's and an SSMEC. Each power supply (A and B) contains overcurrent and overvoltage protection, regulation, and ripple filtering. The dc voltages supplied are +5, +12, and -12. Circuit C is created from unregulated power from power supplies A and B that is OR'ed together and regulated. Power-up detection circuitry is located in the BITE circuitry (see paragraph C.3 of this brief).

Whenever a power bus failure occurs, the power supplies are fed from the remaining bus. Interruption of both supplies for over 100 microseconds will cause the powerdown sequence to occur, shutting down the EIU until power is restored.

The maximum power consumption of the EIU at nominal main dc voltages, will be less than 50.6 Watts for all normal operating modes. Maximum power consumption occurs during MIA transmissions.

TABLE 1.11-II.- POWER SUPPLY DISTRIBUTION

Power supply distribution to EIU's			
EIU	Power and control buses		
1	MNA APC4	MNB APC5	
2		MNB APC5	MNC APC6
3	MNA APC4		MNC APC6
1	CNTAB2	CNTCA2	
2		CNTBC1	CNTAB3
3	CNTBC1		CNTCA2
Power supply distribution within EIU's			
	Power supply		
	A	B	A or B
		Source	
EIU logic description	A	B	C
MIA 1 interface assembly	X		
MIA 2 interface assembly		X	
MIA 3 interface assembly	X		
MIA 4 interface assembly		X	
CIA 1	X		
CIA 2		X	
CIA 3			X
Status buffer 1	X		
Status buffer 2		X	
OI buffer	X		
OI control	X		
OI BITE	X		
GPC BITE			X

F. Operational interface element (OIE) C-G Z 1-5

1. OIE BITE shift register F Z 3-4

The 16-bit OIE BITE register provides storage for the OIE-oriented BITE error signals. The register contents are serially transmitted to the OIE multiplexer to be included as part of the 132-word EIU-to-OIE message. The OIE includes MIA fail flags, CIA fail flags, power fail flags, secondary data flag, repeat flag, OIE buffer fail flag, and the OIE control fail flag. See table 1.11-III for a breakdown of the OIE BITE data word format.

TABLE 1.11-III.- OIE BITE WORD BREAKDOWN

Bit times	OIE BITE word
1	MIA no. 1 Flag
2	MIA no. 2 Flag
3	MIA no. 3 Flag
4	MIA no. 4 Flag
5	CIA no. 1 Flag
6	CIA no. 2 Flag
7	Power supply A fault
8	Power supply B fault
9	Power supply A or B fault
10	S (secondary)
11	R (repeat)
12	B (OIE buffer fail flag)
13	C (OIE control fail flag)
14	Logic "0"
15	Logic "0"
16	Logic "0"

2. Status override detector and status data select G Z 1-3

The status data select performs the function of passing either primary or secondary status channel data to the OIE buffer selection/status register/reset control. The normal selection will be the primary channel. The selection is changed to the secondary channel by the "status override" execute command. The status override detector monitors for any of the MIA's status override commands to be set and sets the S input to the status data select which causes the data switchover. The selection is returned to normal by the "master reset" execute command or the power-on reset signal.

3. OIE Buffer selection/status register/reset control E-F Z 3-4

The OIE buffer selection/status register/reset control provides a double-buffered serial-to-serial transfer point for two blocks of

128 16-bit words of SSMEC status or dump data. The messages, arriving on either the primary or secondary status channel, are loaded into the status register at a 1 MHz rate. Alternate messages are loaded into alternate buffers (A and B) and require about 2.3 milliseconds per message loaded. However, the SSMEC starts message transmission every 40 +/- 2 milliseconds except during memory loading.

Whenever a buffer is completely loaded, the VDT message (together with the two EIU identification words, BITE word, and column parity word) is transmitted to the three OIE output drivers at a 60 kHz rate. Each EIU-to-OIE message requires 35.2 milliseconds for transmission, which means each message is processed by the EIU over a 37.5-millisecond period.

Because of the asynchronous operation of the receive-transmit process, data protection logic is included in the OIE buffer selection/status register/reset control. If the SSMEC memory loading function has caused a status message transmission to the EIU to begin before all previous data blocks have been sent on to the OIE, the EIU logic discards the new data by asserting the reset signal. This prevents the loss of data in the OIE buffer.

If the EIU has not received a new message from the SSMEC, the last message is repeated at completion of the transmission of a message to the OIE and the repeat flag is set. When the alternate buffer is filled, the repeating message is terminated at the end of the block and transmission of the new message is begun and the R flag is cleared.

4. Identification word registers D Z 1-2

The identification word registers are two 16-bit parallel in/serial out registers used by the EIU to insert 32 bits of identification at the beginning of each EIU-to-OIE message. The first 24 bits form a fixed sync pattern. The remaining 8 bits contain the word count of the message received from the SSMEC. See figure 1.11-10 for the OIE ID word formats.

BIT times

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	
111			110			101			111			001			1	First word
001			000			00			WORD COUNT							Second word

Figure 1.11-10.- OIE ID word formats.

5. OIE cycle control D Z 3

The 16-bit OIE cycle control provides the EIU-to-OIE message column parity word data. As each word of the message is transmitted to the OIE, the column parity generator maintains an odd parity bit for each bit position of the word. This is accomplished by processing each bit of generator data against each bit of each outgoing word, with the generator containing zeros at the start of each message.

The column parity word is transmitted to the OIE multiplexer as the last word of the EIU-to-OIE message and does not operate the parity generator.

6. OIE MUX (multiplexer)/manchester encoder/OIE control failure detector and output drivers C-D Z 2-4

The OIE multiplexer provides the logic for combining the 128 VDT words, the two OIE ID words, the OIE BITE word, and the column parity word. The Manchester encoder encodes the serial NRZ bits into Manchester bi-phase code, and drives the three isolated OIE outputs from a single data source at a 60 kHz rate.

As the bits for each word of a message are processed, the OIE column parity generator is signaled so that it can create the column parity word.

The three output drivers are designed to prevent a fault on one circuit from disrupting the message flow on the others.

G. Status buffers (1-2) J-M Z 4-6

1. Status registers 1A, 1B, 2A, and 2B K-L Z 5

   The status registers in the status buffer 1 receive serial data from CIA 1 (primary status data) and subsequently transmit it to MIA 1, 2, or 3 upon receipt of a "response" execute command from the GPC's driving each respective MIA port. Although SSMEC data is available at each of MIA ports 1, 2, and 3, only the GPC attached to MIA 1 issues requests for data ("response" execute commands).

   While data is transferred out of status register 1A, data is loaded into status register 1B and vice versa. The status registers in buffer 2 receive data from CIA 2 (secondary status) and transfer it to MIA 4 upon receipt of a "response" execute command from the GPC driving MIA 4.

   The first 32 words of the 128-word SSMEC status/memory dump data stream are loaded into the status register at the CIA 1-MHz bit rate. At the same time, the validity (E) flag for each word is saved in its E flag bit. When a GPC issues a "response" execute command, the number-of-words field in it determines how many (1 to 32) words are to be transmitted from the status register to the GPC. Responses to requests from MIA 1, 2, or 3 (primary status channel) are available to all three MIA's, but only the requesting MIA can process the data. In practice, only MIA 1 is used to transmit primary SSMEC data. The response to a request from MIA 4 (secondary status channel) is available only to it.

2. Primary status channel/alternate buffer control/MIA/CIA channel timing selection J Z 5-6

   This circuit controls the buffering of data in the two status registers within a status buffer. It tracks when the registers are full or empty and controls when these registers are loaded and emptied.

3. 32-bit E flag shift register/buffer L Z 6

   As each of the 16-bit data words is transferred into the status register from the CIA, the corresponding word valid bit is read into this buffer from the CIA error detection circuitry.

This concludes the discussion of major EIU components. Further description of their interfaces and the flow of data between the SSMEC's and GPC's may be found in systems brief 1.13.

1.12. SSME CREW CONTROLS

1.12.1 General

The crew control of the main engines is limited to meters, status lights, and cockpit switches. This systems brief will describe these crew controls. A discussion of switch redundancy management (RM) is also included.

1.12.2 Switch RM

Most of the switches onboard the Orbiter are designed to be fail-operational/fail-safe. This means that the first failure on a switch will allow its associated function to continue to operate. The second failure will disable the function but in a safe manner. This philosophy is instituted by designing switches with either a voting logic on the contacts or with redundant circuitry. For example, the SSME shutdown pushbuttons and the MPS helium B isolation switches are dual-contact switches. The fail-op/fail-safe criteria are established on the helium B isolation switches by the use of redundant FA MDM's and power supplies. The shutdown pushbuttons are made fail-op/fail-safe by the use of software voting logic on the two discretes coming from the switch. Software voting logic is used only on switches that interface with operational GPC software. Voting logic is called switch redundancy management and is designed into the GN&C switch RM software.

Dual-contact switches that interface with the SSME's and their associated GPC software include the three main engine shutdown pushbuttons, MPS propellant dump sequence switch, and the MPS propellant dump backup LH_2 valve switch. The dual-contact switches require that both discretes be recognized before the function is initiated. Each discrete is set by a contact and is transmitted to the GPC's via flight-critical forward (FF) MDM's. The loss of a contact will disable the function. The loss of an MDM associated with the lost contact will be recognized by the RM software and will remove that discrete from the voting logic. This faults the discrete or command which passes from the contact through the MDM. The remaining discrete and the bypassed MDM are usually sufficient to cause the voting logic to initiate the function. Certain switches such as the SSME shutdown pushbuttons may have special RM software. These switches will be discussed on an individual basis.

Switch RM also exists on certain three-contact switches. On these switches, two of the three discretes must be recognized before the function is initiated. A failure of two of the discretes would disable the function. Table 1.12-I lists the two- and three-contact switches that have switch RM software. The section following table 1.12-I discusses all of the switches, meters, and status lights that the crew can use to operate the main engines.

TABLE 1.12-I.- CONTROLS MONITORED BY GN&C SWITCH RM

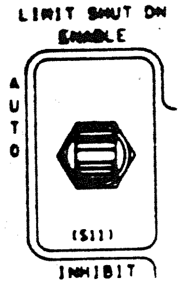
Control	Position	RM level (contacts)	Input parameters	MDM	MML no.
MAIN ENGINE LIMIT SHUT DN switch	ENABLE	3	LIM SHDN ENA CMD A LIM SHDN ENA CMD B LIM SHDN ENA CMD C	FF2 FF3 FF4	V72K0051X V72K0052X V72K0053X
	AUTO	3	LIM SHDN AUTO CMD A LIM SHDN AUTO CMD B LIM SHDN AUTO CMD C	FF2 FF3 FF4	V72K0071X V72K0072X V72K0073X
	INHIBIT	3	LIM SHDN INH CMD A LIM SHDN INH CMD B LIM SHDN INH CMD C	FF2 FF3 FF4	V72K0061X V72K0062X V72K0063X
MAIN ENGINE SHUT DOWN pb - CTR	DEPRESS	2	C SHUTDOWN CMD A C SHUTDOWN CMD B	FF1 FF2	V72K0091X V72K0092X
MAIN ENGINE SHUT DOWN pb - LEFT	DEPRESS	2	L SHUTDOWN CMD A L SHUTDOWN CMD B	FF2 FF3	V72K0093X V72K0094X
MAIN ENGINE SHUT DOWN pb - RIGHT	DEPRESS	2	R SHUTDOWN CMD A R SHUTDOWN CMD B	FF3 FF4	V72K0095X V72K0096X
MPS PRPLT DUMP SEQUENCE switch	START	2	DUMP START CMD A DUMP START CMD B	FF1 FF2	V72K0081X V72K0082X
	STOP	2	DUMP STOP CMD A DUMP STOP CMD B	FF1 FF2	V72K0083X V72K0084X
MPS PRPLT DUMP BACKUP LH_2 VLV - switch	OPEN	2	OPEN CMD A OPEN CMD B	FF3 FF4	V72K0085X V72K0086X
	CLOSE	2	CLOSE CMD A CLOSE CMD B	FF3 FF4	V72K0087X V72K0088X

Component Name: MAIN ENGINE LIMIT SHUT DN switch

Panel Location: C3

Component No.: S11

Configuration and Nomenclature:



Function:

In the INHIBIT position, the GPC issues a limit shutdown inhibit command to each main engine controller. On receipt of the command, the controller inhibits its internal limit shutdown logic.

In the AUTO position, the GPC automatically performs the inhibit function (as described above) on the remaining two engines after one engine shuts down prematurely during mainstage operations or after a data path failure occurs (ref. SCP 2.1.3).

In the ENABLE position, the GPC issues a limit shutdown enable command to each controller. The GPC cannot issue the inhibit command while in the ENABLE position. The controller, on receipt of the enable command, enables its internal limit shutdown logic.

Nominal Usage:

None. Switch is in AUTO.

Contingency Usage:

The INHIBIT position is used to prevent a redline shutdown of an engine while flying through abort gaps and during those periods when the loss of a good engine will cause a nonintact abort.

The ENABLE position is used for enabling limit shutdown on an engine previously inhibited due to loss of data on another engine or a premature engine shutdown. For the case with an engine out and limits inhibited on the remaining two engines, ENABLE is used to enable limit shutdown on the two engines after achieving intact abort capability. The booster engineer will normally ask the flight director to enable limits at the times specified in SCP's 2.1.3 and 2.1.4.

The ENABLE position is also used for low LH_2 NPSP cases. Refer to Flight Rule 5-50, SCP 3.2.6, and systems brief 3.4.

SSME Limit Switch Software

Overview

The SSME OPS sequence monitors the SSME limit switch position and the SSME's for shutdown and for data path failures. The SSME OPS interfaces with the main engines via the SSME SOP sequence. If an SSME is detected as having failed or as having a data path failure, the redline shutdown logic will be automatically inhibited on the other two engines. The inhibit command flags for the "good" engines will be sent by the SSME OPS to the SSME SOP which in turn will send the inhibit commands to the engines. These flags and commands will occur for the first three passes after the engine failure or data path failure. The SSME fail flag or data path fail flag will allow the software to set the inhibit command flags for the other two engines. After the third pass the SSME limits will remain inhibited.

The redline limits on the two good engines are reenabled by placing the limit shutdown switch in the ENABLE, then the AUTO position (refer to SCP 2.1.4). Placing the switch in the ENABLE position results in the SSME OPS sending the limit enable flags to the SSME SOP and the SSME SOP sending the commands to enable the limits. Moving the switch to the AUTO position results in the failed, or data path failed SSME sending out inhibit commands to the other two engines while the good engines are "sending" out their own enable commands. The commands are sent out for the first three cycles, the inhibit command has the higher priority and is read first, then the enable which results in the limits being enabled when leaving the SSME SOP. On the final cycle the last command received is the enable command.

SSME Operations Sequence Details

While reading this section refer to the SSME Operations Sequence documentation in the Booster Systems Software Handbook Section 1.12 and the Redundant Computer Set Logic Flow Diagrams (Lockheed Flows).

The automatic limit sensing is enabled at liftoff with the SSME limit switch is in the AUTO position. For an SSME failure or data path failure the limits on the remaining two engines will be automatically inhibited. To resume automatic limit sensing, the limit switch will be placed in the ENABLE, then AUTO position.

The SSME OPS sequence reads the position of the SSME limit switch. If the switch is in AUTO, counters A (ENABLE position counter) and B (INHIBIT position counter) will be set to zero every pass. The first three passes set the enable flags which are then read by the SSME SOP to send the enable command to the engines. Internal flags C, E, and G are incremented by one on each of the first three passes to allow the enable flags for the center, left and right engines to be set for use by the SSME SOP. After the third pass, the enable flags are no longer set and the limits remain enabled as long as the conditions do not change (i.e., the limit switch is in AUTO and the SSME's are operating with good data paths).

If a data path failure or SSME failure occurs, the limits on the other two engines are automatically inhibited. There are internal counters D, F, and H for the center, left, and right engines, respectively. The failed engine's internal counter is incremented by one on each of the first three passes. This counter controls the setting of the inhibit flag on the other two engines. The SSME OPS will set the inhibit flag for the first three passes after the failure which is used by the SSME SOP to inhibit the limits on the other two engines. This internal counter and its interface with the SSME SOP is the mechanism for inhibiting SSME limits after an engine failure or a data path failure.

With the limits inhibited on the two good engines, the limits can be enabled and automatic limit sensing capability restored. The limit shutdown switch is placed in the ENABLE, then the AUTO position per Flight Rule 5-27. When the limit shutdown switch is in the ENABLE position, the enable flag is set for each engine and the counters C, D, E, F, G and H are set to zero. On each of the first three passes counter A is incremented by one. The enable flag is set on these three passes. The switch is then placed in the AUTO position where the counters A and B are once again set to zero. The engine that failed or has a data path failure will set the inhibit flags for the other two engines via the SSME SOP (as discussed earlier) and the two good SSME's will set their own enable flag. This will occur for the first three passes. The SSME SOP will read both the inhibit and enable flags for the two good engines. Because the inhibit flags have a higher priority, they are processed first. On each pass, the inhibit command is issued first, then the enable command is issued last. On the third and final passes, the final commands issued are the enable commands for the two good engines. Hence, the limits are enabled and automatic limit sensing is enabled.

As a note of interest, after the SSME shutdown limits have been inhibited automatically by the GPC's because of an engine failure or data path failure, positioning the limit shutdown switch to INHIBIT, then AUTO has the same effect as ENABLE then AUTO. INHIBIT-AUTO subjects all the counters to the same logic as the ENABLE-AUTO procedure in the SSME Ops sequence.

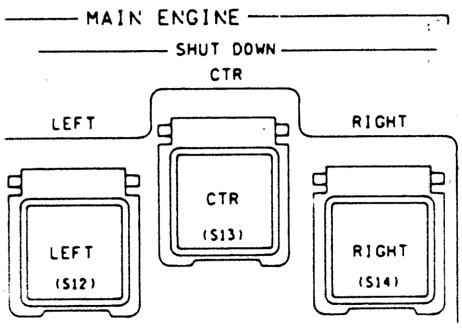
Therefore, reenabling automatic shutdown capability after limits have been inhibited can be accomplished be taking the limit switch to ENABLE, then AUTO or by taking the limit switch directly to the AUTO position if the switch was manually placed in the ENABLE or INHIBIT position. The former procedure is the accepted method and is documented in the Flight Rules.

Component Name: MAIN ENGINE SHUT DOWN pushbuttons

Panel Location: C3

Component No.: S12, S13, S14

Configuration and Nomenclature:



Function:

Each main engine has associated with it a guarded shutdown pushbutton which is located on panel C3. These pushbuttons are dual-contact switches. Associated with each switch contact is a shutdown command and a safing command. Figure 1.12-1 shows the switch and software configuration for the center engine. These switches are used in the following applications:

a. Each switch provides manual shutdown capability if both contacts are functioning.

b. If an engine shuts down and has previously experienced a data path failure, depressing the shutdown pushbutton will send the safing command to the SSME OPS software. This software will set the engine fail flag which will tell guidance that the engine has failed and will close the prevalves on that engine after an appropriate time delay.

c. Backup MECO confirmed can be set by pushing all three buttons simultaneously. This feature is used when data path failures occur on two engines before MECO.

A discussion of the main engine shutdown switch RM and the intricacies of each function will be presented in the following paragraphs.

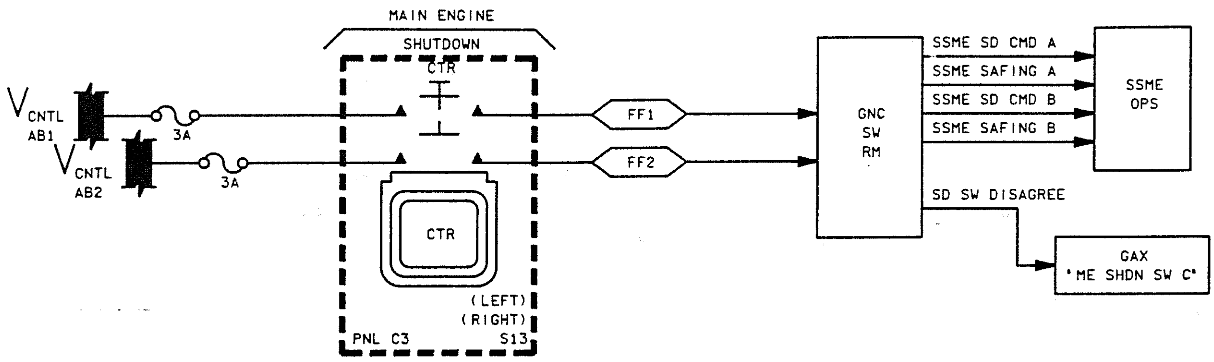


Figure 1.12-1.- SSME shutdown switch.

Main Engine Shutdown Switch RM

If two contacts are available, the output shall be the logical "and" of the two input values. If a failure is detected on one contact for three consecutive Fault Detection Identification/Isolation Recovery/Recognition (FDIR) cycles on a two-contact input, then a contact disagree condition shall be declared. The shutdown command will be output true (shutdown) only if no commfaults exist on either or both of the two associated contact inputs and no contact disagrees have been latched. Thus, protection is provided from an inadvertent SSME shutdown when a switch contact is shorted across its poles, and a commfault then occurs on the good channel.

If a data path failure has occurred, and the engine has shutdown prematurely, depressing the pushbutton will send the safing command. The safing command is used by the SSME OPS to set the engine fail flag and close the associated prevalves. If one contact has been commfaulted, the shutdown function is lost but the safing command will still be available.

If power is not available because of a bus failure or an open switch contact failure, neither the manual shutdown commands nor the safing commands will have any affect. Neither manual shutdown nor guidance moding can occur. If the pushbutton is depressed, a class 3 alert, ME SHDN SW L (C,R), will warn the crew that the pushbutton is no longer functioning.

Backup MECO Confirmed

CR39091E was approved for Version 18 (PASS) and Tape 11 (BFS) software for STS-2 on February 19, 1981. The external tank separation sequence is entered when the MECO-confirmed flag is set in the SSME OPS sequence. The MECO-confirmed flag is automatically set by the software when the GPC's see all three SSME chamber pressures are less than 30 percent or if two of three Pc's are less than 30 percent and one engine has a data path failure. MECO-confirmed will not be automatically set if two data path failures occur. Crew action is required to set a backup MECO-confirm for moding into the ET separation sequence.

During STS-1, the crew would use the SPEC 51 override display and OPS 104 PRO when trying to mode to 104. Also for STS-1, the backup ET SEP AUTO item on the override display would set MECO-confirmed for RTLS cases. Unfortunately, if the backup ET SEP AUTO item was used before RTLS selection, it would have shut the engines down prematurely. To prevent this from occurring, CR39091E changed the SSME OPS for STS-2 so that the backup ET SEP AUTO item does not set the MECO-confirmed flag. In its place, CR39091E set the MECO-command and MECO-confirmed flags when the safing and manual shutdown commands are set by pushing all three shutdown pushbuttons. Due to an oversight in the PASS (Version 18) software, the safing commands were not latched when the button was pushed. The button had to be held down for the GPC to see the safing command. For this reason, the crew held all three pushbuttons down together for at least 1 second so that all three engine safing commands could be read simultaneously. This procedure was to be used for nominal, AOA, ATO, RTLS, and TAL cases. The OPS 104 PRO procedure still works for nominal, AOA, and ATO. It should be noted that a failed open contact or a failed control bus will prevent the backup MECO-confirm from being set with the shutdown pushbuttons. The OPS 104 PRO procedure would then be used.

Since the prevalves close when the safing command is recognized, the crew under the Version 18 software would have to hold a shutdown pushbutton down for 9.4 seconds to allow the LO2 and LH2 prevalve timers to time out. Prevalve closure is not essential to a safe pre-MECO engine shutdown; therefore, the crew was advised to hold the button down for 1 second. Prevalve closure would occur at MECO when the MECO-command flag was set. This problem with latching the safing commands was fixed for STS 41-D.

Latch Main Engine Safing Commands

CR39430 was approved for Version 21 (PASS STS 41-D). The safing command is latched in the SSME OPS sequence so that backup MECO-confirmed can be set by pushing each shutdown button one at a time. Also a shutdown button need not be held for 9.4 seconds to close the prevalves on an engine.

Table 1.12-II summarizes the software and functional differences as discussed above.

TABLE 1.12-II.- SSME SHUTDOWN PUSHBUTTON LOGIC

Flight	Function \ Switch	Two contacts available	One contact available One contact shorted (switch disagree) (1)	One contact available One MDM commfault (2)
STS-1	Manual shutdown	PASS & BFS	PASS & BFS One more subsequent contact short would shut down an engine	PASS One more subsequent contact short would shut down an engine
	Mode Guid. for eng out with data path failure	PASS & BFS	PASS & BFS	PASS
	Backup MECO confirmed			
	Prevalve closure with engine out	PASS & BFS	PASS & BFS	PASS
STS 2-11	Manual shutdown	PASS & BFS	PASS & BFS No man SD command class 3 crew alert and message "ME SHDN SW (L(C,R)"	No man SD command
	Mode Guid. for eng out with data path failure	PASS & BFS	PASS & BFS	PASS
	Backup MECO confirmed	PASS & BFS Hold three SD pb's simultaneously for 1 sec	PASS & BFS Hold three SD pb's simultaneously for 1 sec	PASS Hold three SD pb's simultaneously for 1 sec
	Prevalve closure with eng out	PASS & BFS Hold pb for 9.4 sec	PASS & BFS Hold pb for 9.4 sec	PASS Hold pb for 9.4 sec
STS 41-D and subs	Manual shutdown	PASS BFS	No man SD command class 3 crew alert and message "ME SHDN SW (L(C,R)"	No man SD command
	Mode Guid. for eng out with data path failure	PASS BFS	PASS BFS	PASS
	Backup MECO confirmed	PASS BFS	PASS BFS	PASS
	Prevalve closure with eng out	PASS BFS	PASS BFS	PASS

(1) One contact failed open, due to either a failed open switch or a low control bus voltage, will completely preclude the operation of the manual shutdown pushbutton (i.e., no manual shutdown, no moding guidance for an engine out with a data path failure, and no setting the backup MECO-confirm logic).

(2) There is no MDM commfault protection in the BFS.

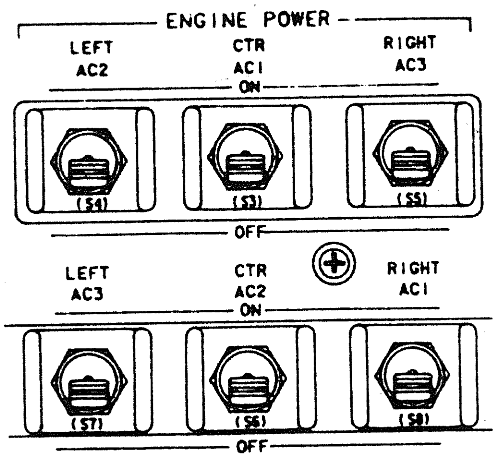
Component Name: MPS ENGINE POWER switches

Panel Location: R2

Component No.: S4, S3, S5, S7, S6, S8

Note: AC OA, B, C, ENGINE C, L, R circuit breakers (panel L4) provide circuit protection (18 cb's).

Configuration and Nomenclature:



Function:

These switches turn MPS engine controller power ON and OFF. There are two ac power supplies to each engine controller, and the engine will operate with either supply. There is no computer or ground control for engine power.

Nominal Usage:

The switches will be in the ON position from prior to the start of the countdown until the completion of the propellant dump when the power will be turned off.

Contingency Usage:

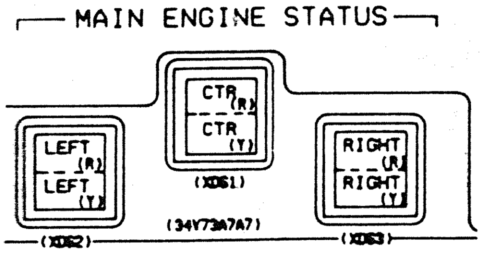
Positioning both power switches for an engine to OFF while the engine is running will cause the engine to shut down pneumatically. Therefore, the switches provide a means of shutting down an engine if the normal shutdown command path has failed. For significant Pc sensor shifts and corresponding mixture ratio shifts, the affected controller channel will be turned off to disqualify the failed sensors and return the engine to nominal performance. This procedure is only used when the predicted vehicle performance indicates that an ATO cannot be achieved and no other engine problem exists that will cause an engine shutdown or degrade engine performance.

Component Name: MAIN ENGINE STATUS lights

Panel Location: F7

Component No.: XDS1, XDS2, XDS3

Configuration and Nomenclature:



Function:

The red light is an indication of any main engine shutdown (provided that controller data is valid) or that an engine limit is exceeded on both channels as determined by the controller. The amber light is an indication of the following engine failure conditions:

- Electrical lockup
- Hydraulic lockup
- Data path failure
- Command path failure

The crew will also receive a corresponding fault summary message for each of these four failures.

Nominal Usage:

The illumination of the three red lights indicates MECO. These red lights extinguish at external tank separation.

Contingency Usage:

Red light - This light indicates engine out and is one of the primary cues for taking abort action. It is also an indication of a limit-exceeded problem on an engine any time the shutdown limits are inhibited.

Amber light - For hydraulic lockup or electrical lockup, this light will prepare the crew for an impending abort gap due to an engine locked in the max q thrust bucket. For dual failures above RPL, the 3g limit will be violated. The nongimbaling engine will be shut down prior to MECO if two hydraulic lockup conditions are caused by APU failures.

For an engine with a data path failure, the amber light may require the crew to provide engine out information to the GPC and an early manual shutdown near MECO by pushing the main engine shutdown pushbutton for that engine.

For an engine with a command path failure, the amber light indicates that a manual shutdown using the main engine power switches will be required on that engine.

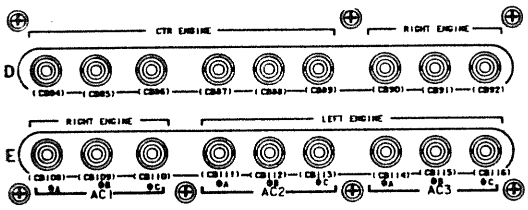
Component Name: ENGINE AC circuit breakers

Panel Location: L4

Component No.:

cb 84, 85, 86, 87, 88, 89,
90, 91, 92, 108, 109, 110,
111, 112, 113, 114, 115, 116

Configuration and Nomenclature:



Function:

Provide circuit protection for main engine controllers. Loss of a single phase causes loss of the associated controller channel (i.e., loss of controller redundancy). The loss of a single phase on two ac buses will shut down a main engine.

Nominal Usage:

None. Will be "set" on launch pad through the entire flight.

Contingency Usage:

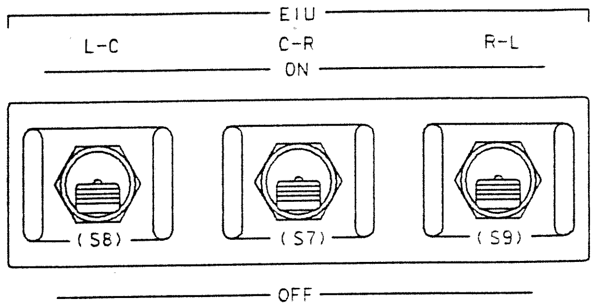
Detection of high circuit load.

Component Name: EIU

Panel Location: 017

Component No.: S8, S7, S9

Configuration and Nomenclature:



Function:

These switches provide power to or remove power from the EIU's. This is the only power supplied to the EIU's. The loss of power to an EIU would cause a data path and command path failure on its associated SSME.

Nominal Usage:

These switches are positioned to ON during prelaunch countdown and are positioned to OFF at a Mission Elapsed Time (MET) of approximately 1 hour.

Contingency Usage:

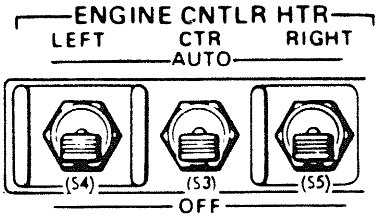
None.

Component Name: MPS ENGINE CNTLR HTR switches

Panel Location: R4

Component No.: S4, S3, S5

Configuration and Nomenclature:



Function:

These switches control the dc power supply to the main engine controller heaters. These heaters provide thermal conditioning for the controller during on-orbit operations. In the AUTO position, power is supplied to the controller thermostat, which maintains the controller temperature between 10 to 20° F. There is no computer or ground control for the controller heaters.

Nominal Usage:

The switches will be in the OFF position for launch and will be manually positioned to AUTO on orbit if the controller temperature drops below -50° F. This switch can be left in the OFF position since the temperature normally drops to only -30° F and then rises above 0° F.

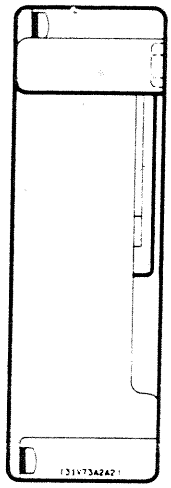
Contingency Usage:

The heaters can be turned OFF to conserve power during a power contingency.

Component Name: Speed Brake Thrust Controller control

Panel Location: C3

Configuration and Nomenclature:



Function:

The manual throttling capability is supported by PASS but not by BFS and is restricted to the pilot's speed brake only. The pilot depresses and holds the right hand speed brake/throttle controller (RH SBTC) takeover button located on the right side of the SBTC handle on panel C3. (The commander cannot take manual throttles.) Depressing the SBTC takeover button extinguishes the AUTO lamp of the SBTC pbi on panels F2 and F4 and freezes the throttle command at its last value before takeover. The pilot moves the SBTC handle forward or aft, with the takeover button still depressed until the takeover manual command matches the last auto throttle command. (If the throttle command is not matched before the takeover button is released, the throttle control will be returned to auto throttling and the AUTO SBTC lamp will be illuminated.) When the manual throttle setting matches the auto throttle setting, the MAN lamp of the RH SBTC pbi will be illuminated. With manual throttling engaged, the SBTC takeover button can be released. Subsequent movements of the SBTC handle will generate corresponding throttle commands. The SBTC-compensated position is converted into a throttle setting in percent rated power level (RPL) and is limited to a value within the minimum (65 percent RPL) and maximum (109 percent RPL) throttle command range. Depressing the takeover button during manual throttling has no effect. To return to auto throttling from manual control, either the LH or RH SBTC pbi (on panels F2 and F4, respectively) is depressed, which illuminates the AUTO lamp. Throttle matching is not required before returning to auto throttle control. If manual throttling is engaged and maintained until MECO, the SSME's must be shut down at MECO with the ac power switches and shutdown pushbuttons.

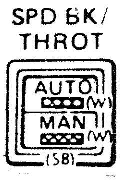
Manual throttling is used in the following cases. If the LH_2 net positive suction pressure (NPSP) is low, the SSME's must be throttled down to prevent pump cavitation. The SSME's will be manually throttled to maintain the minimum required LH_2 NPSP at the engine inlet. Reference manual throttle conditions for low LH_2 NPSP in Booster flight rule 5-50. If TAL abort is selected and the time of MECO is not stable in 10 seconds after guidance initiate, manual throttle will be engaged. Manual throttling is also used to protect LO_2 NPSP at engine shutdown for two-engine out cases as described in Booster flight rule 5-38. For contingency aborts, manual throttling is engaged. Reference the manual throttle conditions outlined in the trajectory flight rules 4-22 and 4-29.

Component Name: SPD BK/THROT pushbutton indicators

Panel Location: F2, F4

Component No.: S8

Configuration and Nomenclature:



Function:

The pushbutton is used to change the SSME throttle control from manual to automatic. As an indicator, it identifies the throttle control mode.

The white AUTO light indicates that the throttle control is automatic. The white MAN light indicates that the throttle control is manual.

Nominal Usage:

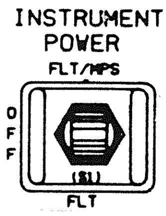
None.

Component Name: INSTRUMENT POWER switch

Panel Location: F6

Component No.: S1

Configuration and Nomenclature:



Function:

In the FLT/MPS position, power is supplied to the Pc meter, the ENG MANF meter, the HELIUM meter on panel F7, and the flight critical dedicated displays on panels F6 and F7.

In the OFF position, power is removed from the dedicated meters and displays.

In the FLT position, power will be supplied to the flight-critical dedicated displays on panels F6 and F7 but will not be supplied to the MPS meters. This can be performed on-orbit to save power.

Nominal Usage:

The switch is in FLT/MPS during ascent.

Contingency Usage:

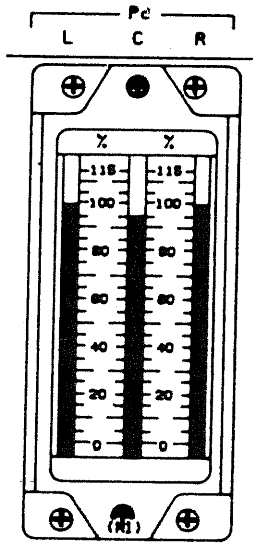
None.

Component Name: MPS PRESS Pc meters

Panel Location: F7

Component No.: M1

Configuration and Nomenclature:



Function:

Tape meters display the main combustion chamber pressure (Pc) of each engine. For each engine, the average of the Pc sensors A and B pairs is computed in the controller and routed to the GPC, which scales it appropriately to drive the meters.

Nominal Usage:

Verify proper engine operation, especially during throttling periods. Verify MECO.

Contingency Usage:

Engine shutdown confirmation for an engine failure condition. For a data path failure, the affected engine Pc meter will be driven to zero. For a stuck throttle condition (hydraulic lockup, electrical lockup, or command path failure), the affected engine Pc meter will remain static and not change with the other two engine Pc meters during engine throttling.

1.13 SSME AVIONICS INTERFACES

1.13.1 General

The exchange of command messages and performance data between the orbiter's GPC's and the SSME's takes place through a network of avionics equipment as depicted in figure 1.13-1. The major components in this network between the GPC's and SSME's are flight-critical data buses, EIU's, SSME controllers, and hydraulic systems.

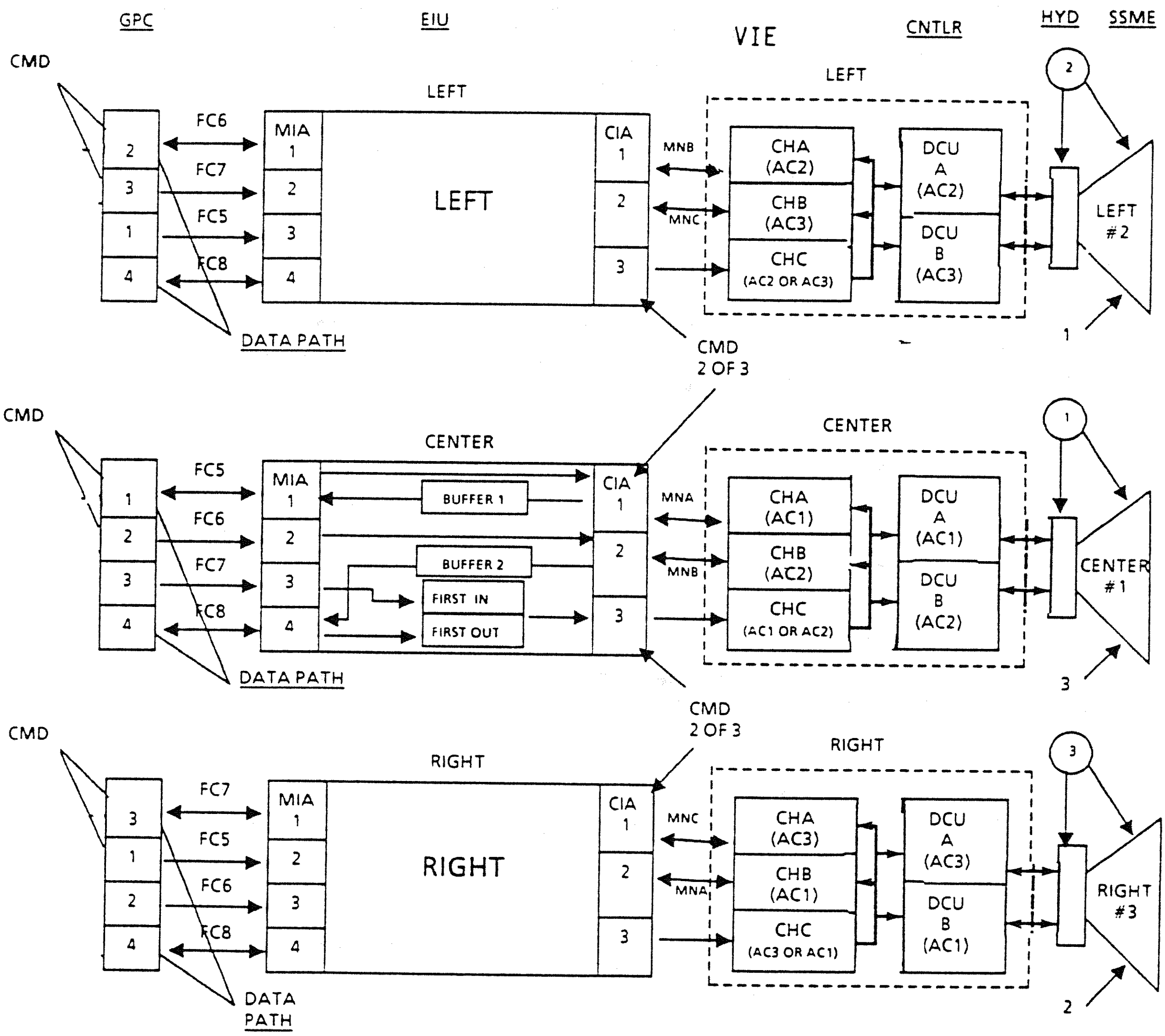


Figure 1.13-1.- SSME avionics interfaces.

1.13.2 Functional Description

1.13.2.1 Command Flow

Each SSME has an EIU, which is independent of the other EIU's, to act as a data transfer interface between the dedicated SSME controller and the GPC's. The EIU uses four MIA ports to interface with the GPC's. Each of the four GPC's in the primary avionics software system (PASS) transmits SSME commands over a flight-critical data bus to a corresponding MIA port. The MIA's check the commands for proper sync, the correct number of bits, and odd parity. Commands that are not validated are dead-ended in the MIA's. The validated commands from MIA's 1 and 2 are sent to controller interface adapters (CIA's) 1 and 2, respectively. The validated commands from MIA's 3 and 4 pass through a first-in first-out logic box. The first valid command that enters the logic box is transmitted to CIA 3; the other is dead-ended. From the CIA's the three validated commands are transmitted to the controller. In the controller, the vehicle interface electronics (VIE) checks for transmission errors and then sends the commands to digital computer unit (DCU) A and B. DCU A is normally in control; DCU B assumes control if DCU A fails. If DCU B subsequently fails, power will be removed from the fail-safe solenoids and a pneumatic shutdown of the SSME occurs.

The DCU that is in control compares the three commands received from the VIE. These commands must pass voting. All types of commands (variable or absolute) must agree exactly bit for bit.

The one variable command is the MAIN CHAMBER PRESSURE LEVEL. This command sets the engine main chamber pressure level, which is variable from 65 percent to 109 percent power level in 1 percent increments. The remaining engine commands are absolute commands such as START ENABLE, START, LIMIT CONTROL INHIBIT, LIMIT CONTROL ENABLE, SHUTDOWN ENABLE, SHUTDOWN, DUMP START, AND DUMP STOP.

If only two of the three commands agree, or if none of them agree, a failure will be reported by changing the engine status word to major component failure and by setting FID 42. No more than one FID 42 will be posted after engine start.

If two of the commands agree, and the commands are not START ENABLE or START, then the vote is defined as successful. If only two of the commands agree, and the commands are START ENABLE or START, the vote is defined as unsuccessful.

After the voting is passed, the DCU channel determines whether the voted command is valid for the phase of engine operation. If the command is valid, the controller will send its own commands to the SSME in response to the GPC-commanded function.

Command path failure results from the failure of the GPC's or flight-critical data buses assigned to MIA's 1 and 2, the failure of MIA's 1 and 2, the failure of any two of the three CIA's, failure of any two of the three

VIE channels, or the failure of any two controller channels. The loss of command capability to an engine will mean that the engine will not respond to throttle commands, shutdown commands, limit inhibit/enable commands, GPC shutdown commands, and MPS dump commands. That engine must be shutdown manually via the ac switches and SD pushbuttons.

If a command is sent and voting is passed but the command is not valid for the current engine opening mode, the controller sends COMMAND REJECTION status. This is received by the GPC's and a command path failure message is annunciated. For example, when the GPC dump start command is sent after an engine has experienced a pneumatic shutdown, a COMMAND REJECTED and a Command Path message will be sent because the command is not valid.

1.13.2.2 Data Flow

The data from the sensors in the SSME's are arranged into a vehicle data table (VDT) in the controller. The controller transmits the VDT, consisting of 128 words, to CIA's 1 and 2. The data transmitted to CIA 1 is called primary data, and the data transmitted to CIA 2 is called secondary data. In the EIU, the first 32 words of the VDT are loaded into a status register for transmission upon request by the GPC's. The remaining words of the VDT are not available real time in the MCC. Upon GPC request, the 32 words are transmitted from MIA 1 and the first 6 words of the 32 words are transmitted from MIA 4. The MIA interface adds a sync pulse, address, and check pattern to each word. The primary data path is through MIA 1, and the secondary data path is through MIA 4.

Loss of CIA 1, MIA 1, VIE channel A, or the associated string (FC data bus and GPC) will cause loss of primary data. Secondary data (PC, ESW, FID, TREF ID WD 1&2) will still be available to monitor engine status.

A data path failure, which is total loss of data from the SSME's to the GPC's, results from the failure of the GPC's or flight-critical data buses assigned to MIA's 1 and 4, the failure of MIA's 1 and 4, the failure of CIA's 1 and 2, the failure of VIE channels A and B, or the failure of controller channels A and B.

The command and data flow for BFS-engaged is the same for PASS-engaged, with the exception that all eight flight-critical data buses are assigned to GPC 5. Only VDT words 1-6 are available in BFS. If two PASS GPC's are lost, the resulting command or data path failure can be recovered by engaging the BFS or by restringing the data buses. Restringing is only done during ascent if the engine with the command path is stuck in the throttle bucket.

1.13.2.3 Power Buses

Each DCU of the controller is powered by a different ac bus, as shown in table 1.13-I.

TABLE 1.13-I.- AC BUS CHART

SSME	DCU channel	AC bus
C	A B	AC1 ϕA,B,C AC2 ϕA,B,C
L	A B	AC2 ϕA,B,C AC3 ϕA,B,C
R	A B	AC3 ϕA,B,C AC1 ϕA,B,C

Each ac bus has three phases. The loss of one or more phases in a bus causes channel failure, resulting in the loss of redundancy in the controller. If no other failures occur, loss of redundancy has no adverse affect on continued SSME performance. If, however, there are additional failures affecting that engine controller, SSME shutdown may occur. In particular, two possible failure scenarios deserve further examination.

A. If a shutdown sensor assigned to the remaining channel fails above or below its shutdown limit, that controller channel will command a hydraulic shutdown of the SSME if the shutdown limits are enabled.

B. If the remaining channel subsequently fails, power will be removed from a fail-safe pneumatic solenoid valve and a pneumatic shutdown of the SSME will occur.

The power interface for each EIU was changed during the post 51-L SDRI review. The orbiter wiring was changed such that a single switch failure will not power down an EIU, which would have resulted in a Command and Data Path Failure on an engine. The current wiring is shown in figure 1.13-2. The power supply to each EIU is single-failure tolerant.

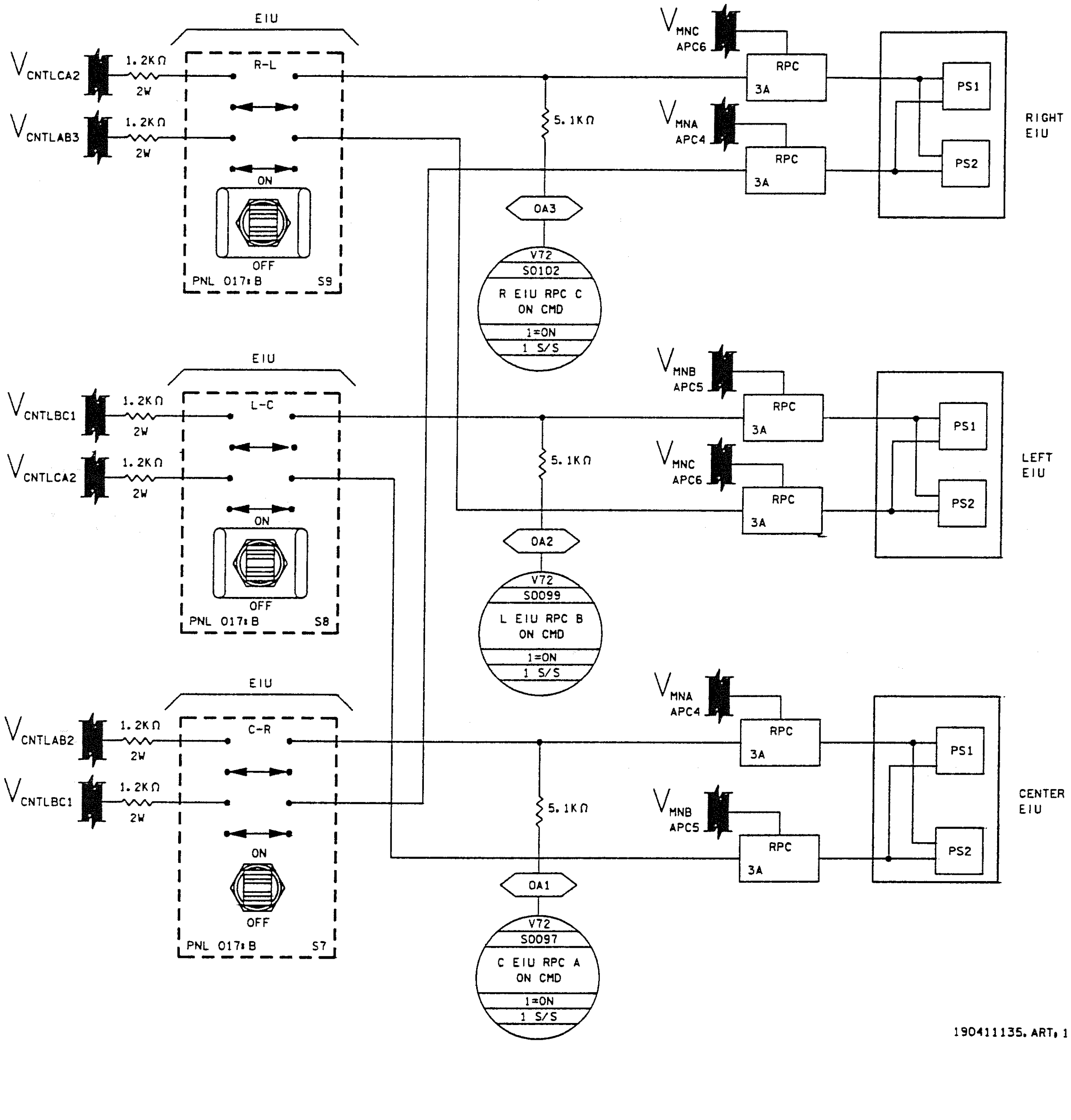


Figure 1.13-2.- Orbiter switch wiring.

1.13.2.4 Hydraulics

Each SSME has one APU to actuate the engine valves. APU's 1, 2, and 3 operate valves on the center, left, and right SSME's, respectively. Therefore, the loss of one APU will result in hydraulic lockup on one SSME.

TABLE 1.13-II.- HYDRAULICS

SSME	TVC (APU's)	Valve (APU's)
C	1 & 3	1
L	1 & 2	2
R	2 & 3	3

As seen above, the failure of two APU's will result in the hydraulic lockup and loss of TVC capability on one SSME, hydraulic lockup and loss of TVC redundancy on another SSME, and loss of TVC redundancy on the third SSME.

1.13.3 References

1. Interface Control Document - 13M15000

2. Functional Subsystem Software Requirements - System Interface

3. Computer Program Contract End Item Spec - CP K06R0001

1.14 SSME CONTROLLER HARDWARE

1.14.1 General

The controller is a single, integral electronics package mounted on the SSME. The controller is specifically designed to operate in conjunction with engine sensors, valves, actuators, spark ignitors, harnesses, and the operational computer program to provide a self-contained system for engine checkout, control, and monitoring. The size of the controller is 23.5 by 14.5 by 17.0 in., and it weighs 211 lb. See figures 1.14-1 through 1.14-5. The controller is packaged in a sealed, pressurized chassis with cooling provided by convection heat transfer through pin fins as part of the main chassis. The electronics are distributed on functional modules having special provisions for thermal and vibrational protection. The main features of the controller are:

a. Design flexibility - The controller is easily reprogrammed to allow for design flexibility and growth potential. The controller's digital computer unit (DCU) allows modification of engine control equations and constants by changing the stored program. The DCU updates instructions to the engine control elements every 20 milliseconds (50 Hz). The DCU uses 16-bit computations, 12-bit input/output resolution, and self calibrating analog-to-digital conversion to achieve precise closed-loop engine control.

b. Onboard checkout - Onboard checkout and monitoring minimizes electronic ground support equipment (GSE). In-flight diagnostic service and engine redline limit monitoring are provided. Data acquisition for operation and maintenance are also provided.

c. Sequencing - The controller verifies that the engine is ready to start. The sequences for starting and shutdown are provided by the controller.

d. Closed loop control - Thrust and mixture ratio are controlled independently by two main valves. The OPOV controls thrust and the FPO controls mixture ratio.

e. Fail-operational/fail-safe design - Engine reliability is enhanced by a dual-redundant control system. The redundancy management design provides fully operational avionics after the first failure. This design ensures a safe engine shutdown after the second failure. High reliability parts are used throughout the controller.

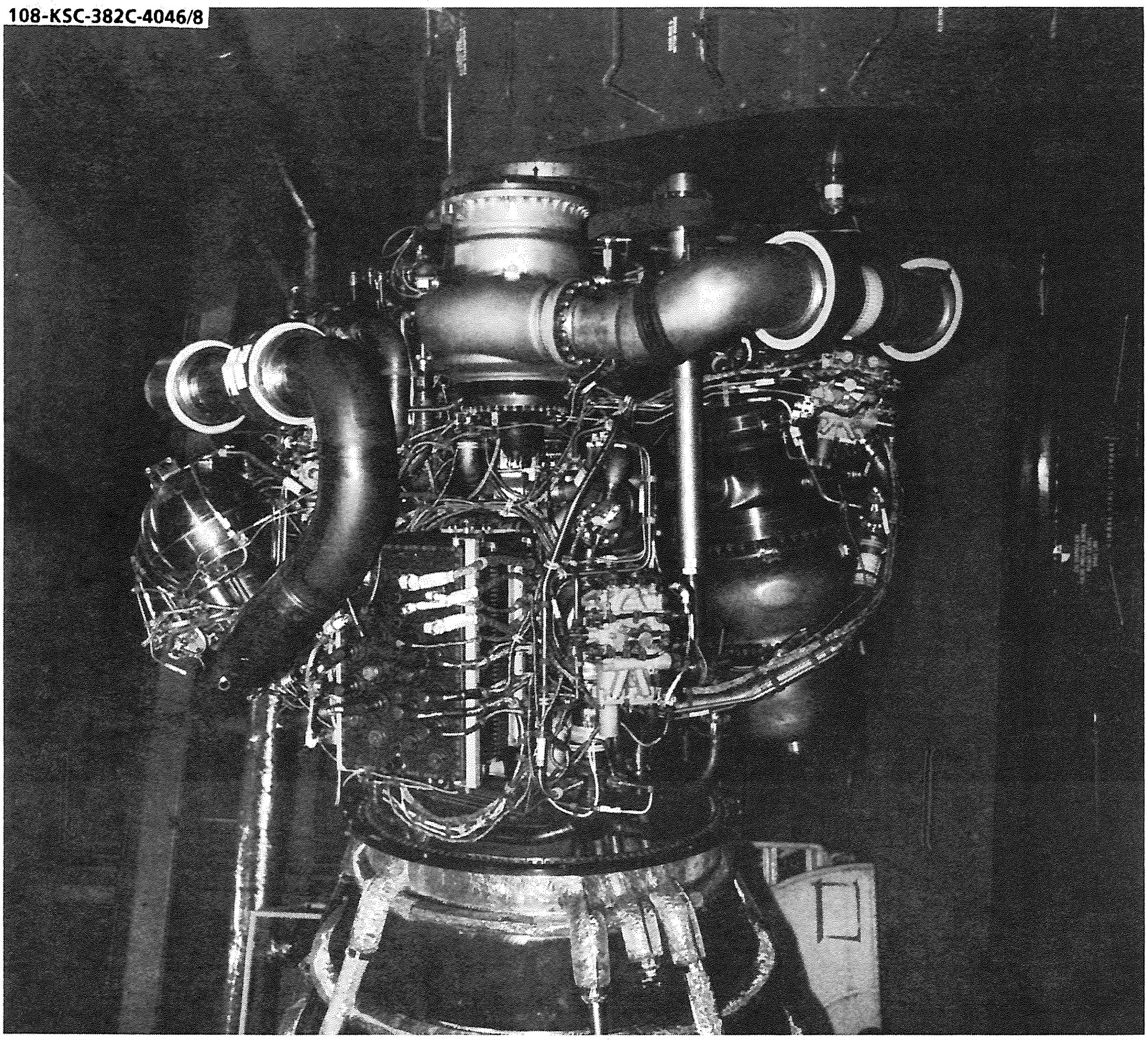


Figure 1.14-1.- SSME-view 1.

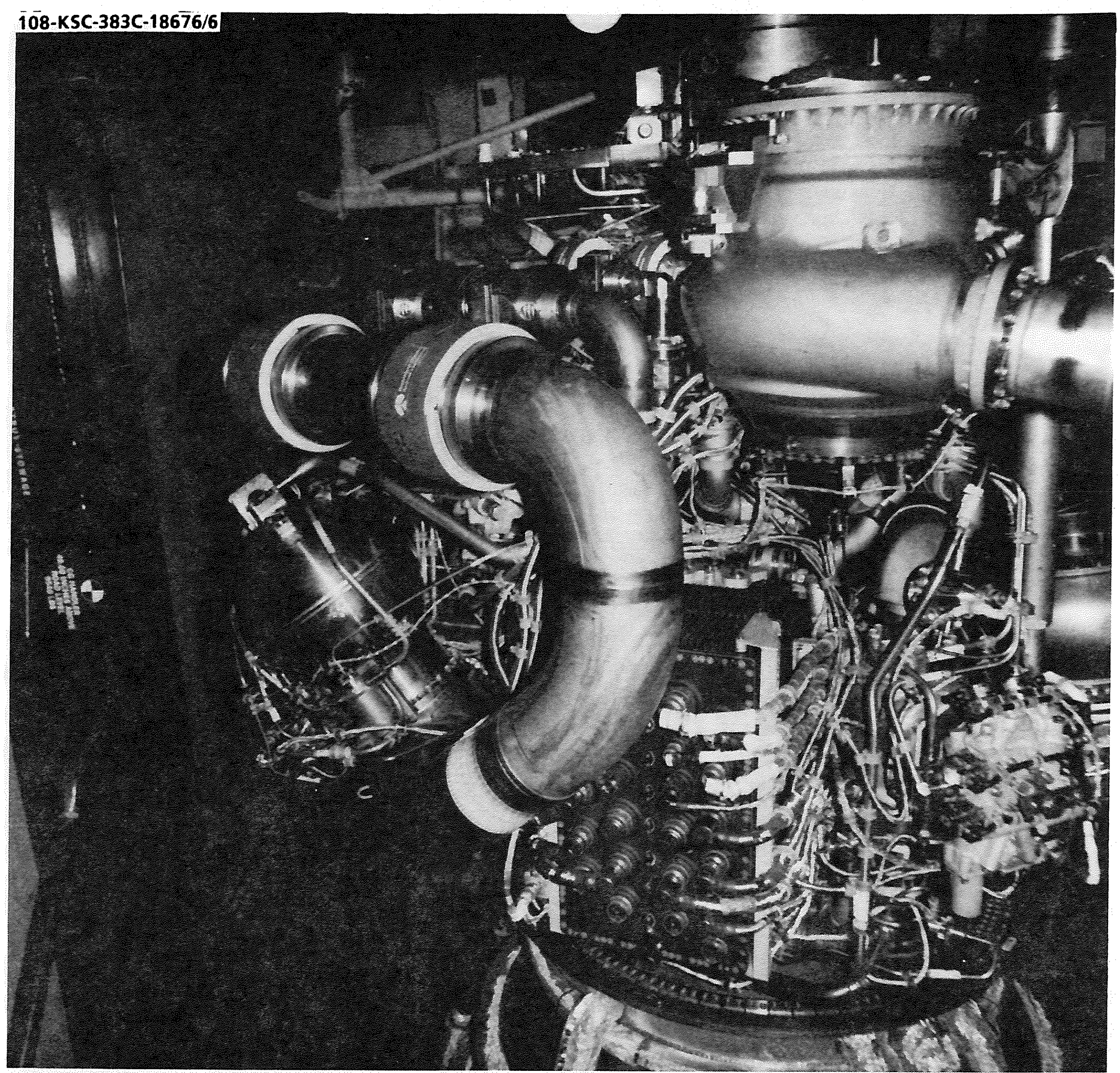


Figure 1.14-2.- SSME-view 2.

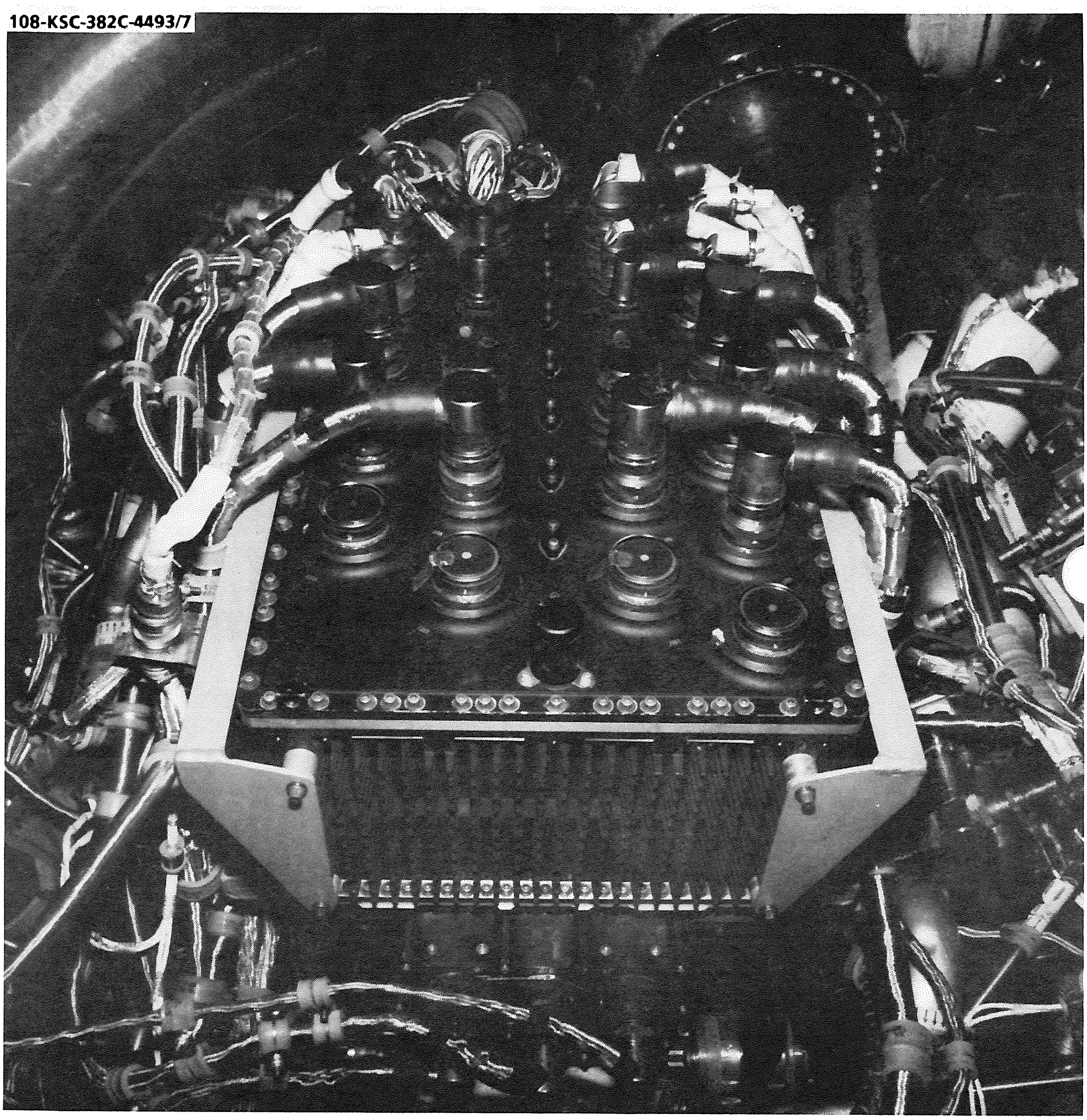


Figure 1.14-3.- SSME-view 3.

* Size	23.5 x 14.5 x 17.0 in.
* Weight	211 lb
* Input power	115 +/- 5 V, 400 Hz, three-phase 28 V dc (heater) Standby: 490 watts Mainstage: 600 watts
* Heat transfer	Ground: Forced air cooling Flight: Convection cooling
* Temperature environment	Operating: -50° to +95° F Non operating: -220° to +200° F
* Vibration environment	Decaying sine: 24.0 G's peak Random: 22.5 G's RMS
* Mounting	Four point vibration isolators

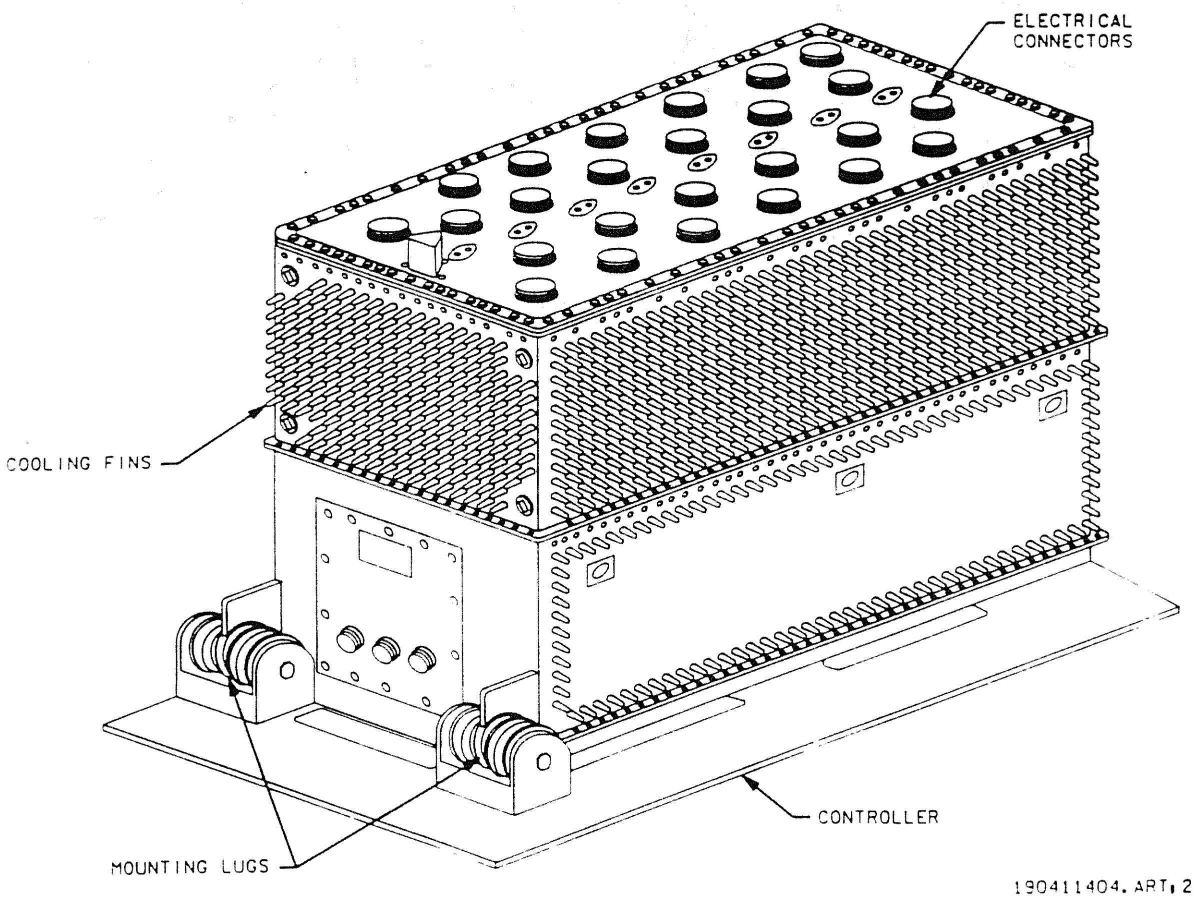


Figure 1.14-4.- Controller physical characteristics.

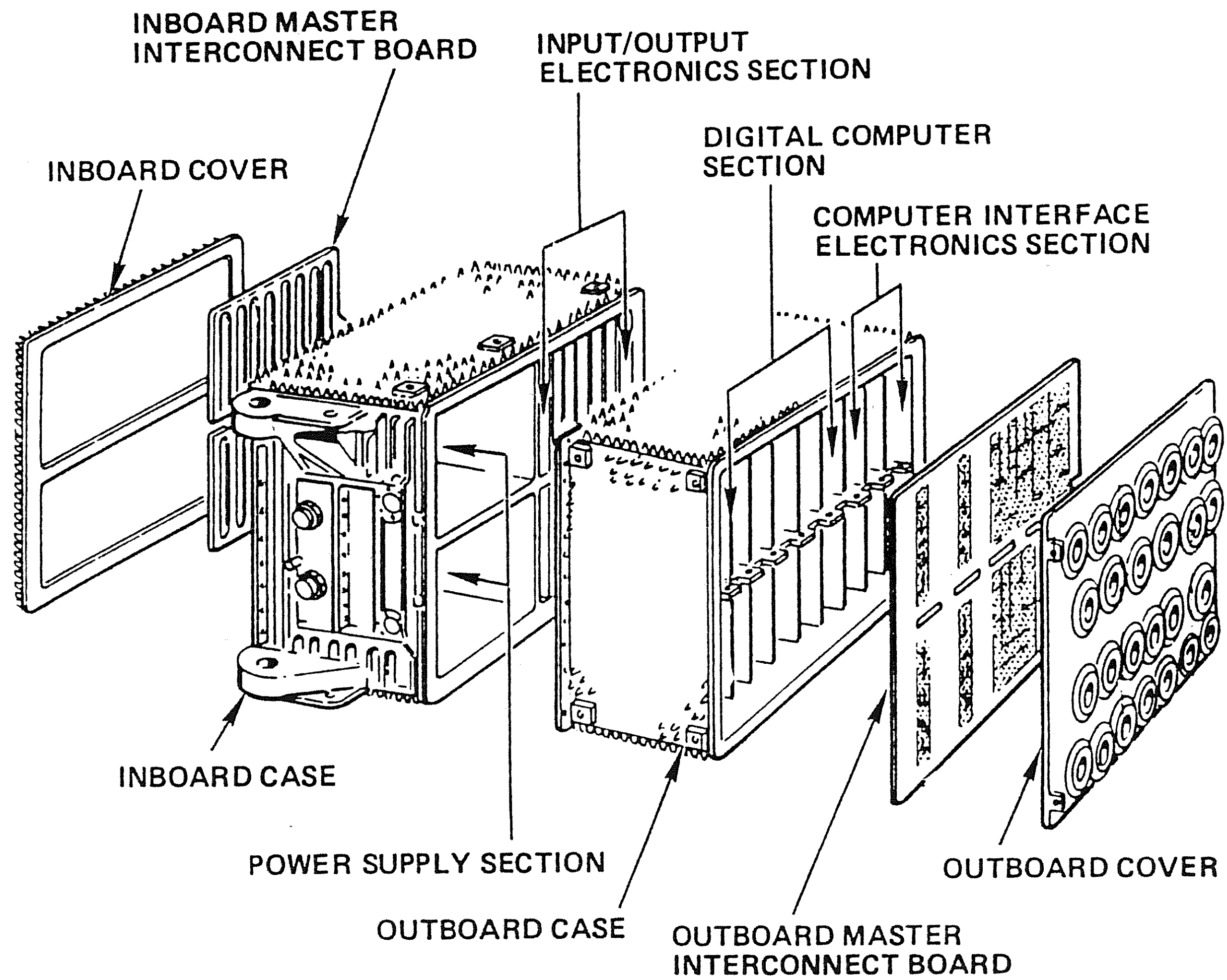


Figure 1.14-5.- SSME controller.

1.14.2 Functional Organization

The controller is divided into five subsystems. Each of the five subsystems is duplicated to provide dual-redundant capability. See figures 1.14-6 and 1.14-7. Reference SSSH drawing 10.4.

1.14.2.1 Input Electronics

The primary purpose of the input electronics (IE) is to acquire raw engine performance data from all engine sensors. Engine control sensors and redline sensors are dual redundant. Maintenance data sensors are not redundant. The IE conditions the signals from the sensors and converts them into digital values. The IE multiplexes that data and transfers it for processing by both DCU's. Each data transfer is initialed by the software, and it is controlled by direct memory access (DMA) control logic in the computer interface electronics (CIE) of the controlling computer. Reference SSSH drawing 10.6.

The input electronics consists of two major sections: the dual-redundant sensor electronics, and the dual-redundant input interface electronics.

1.14.2.1.1 Sensor electronics.- The sensor electronics provides interface with the engine sensors and the necessary sensor channel checkout provisions. The two electronics channels are isolated and obtain their operating voltages from the separate channels in power supply electronics. The excitation for temperature and pressure-sensor bridge networks is provided by separate channels of the special dual-redundant reference power supply.

Sensor electronics Channel A and Channel B outputs are processed by the respective A and B channels in the input interface electronics.

A pair of the four (dual-dual) outputs from the fuel flowmeters is processed by each input electronics channel.

1.14.2.1.2 Input interface electronics.- The balance of the input electronics functions is provided by the dual-redundant input interface electronics channels.

Each channel interfaces with its dedicated computer via the interface buffer. Data is routed to the interface buffer by logic signals which are derived from the information contained in a 8-bit address word received from computer interface electronics associated with the controlling computer.

1.14.2.2 Output Electronics

The output electronics (OE) convert the computer digital control commands into voltages suitable for powering the three types of engine controls: spark ignitors, the on/off solenoid values, and the hydraulic servoactuators for the propellant valves. Reference SSSH drawing 10.7.

The OE consists of two identical channels, each of which is dedicated to an associated channel of spark igniters, solenoid valve coils, servovalves, and actuator position transducers. Either OE channel is capable of operation with either CIE channel. A watchdog timer signal, received from the CIE channel, determines which CIE channel is controlling the OE. Each OE channel receives electrical power from its own power supply electronics channel. The OE receives and thus decodes commands issued by the CIE and thus determines which igniter, solenoid coil, or serovalve is to receive the command. The OE also provides sensor checkout control signals to the IE, contains provision for analog closed-loop control of actuator position, and provides built-in test data to the CIE and IE.

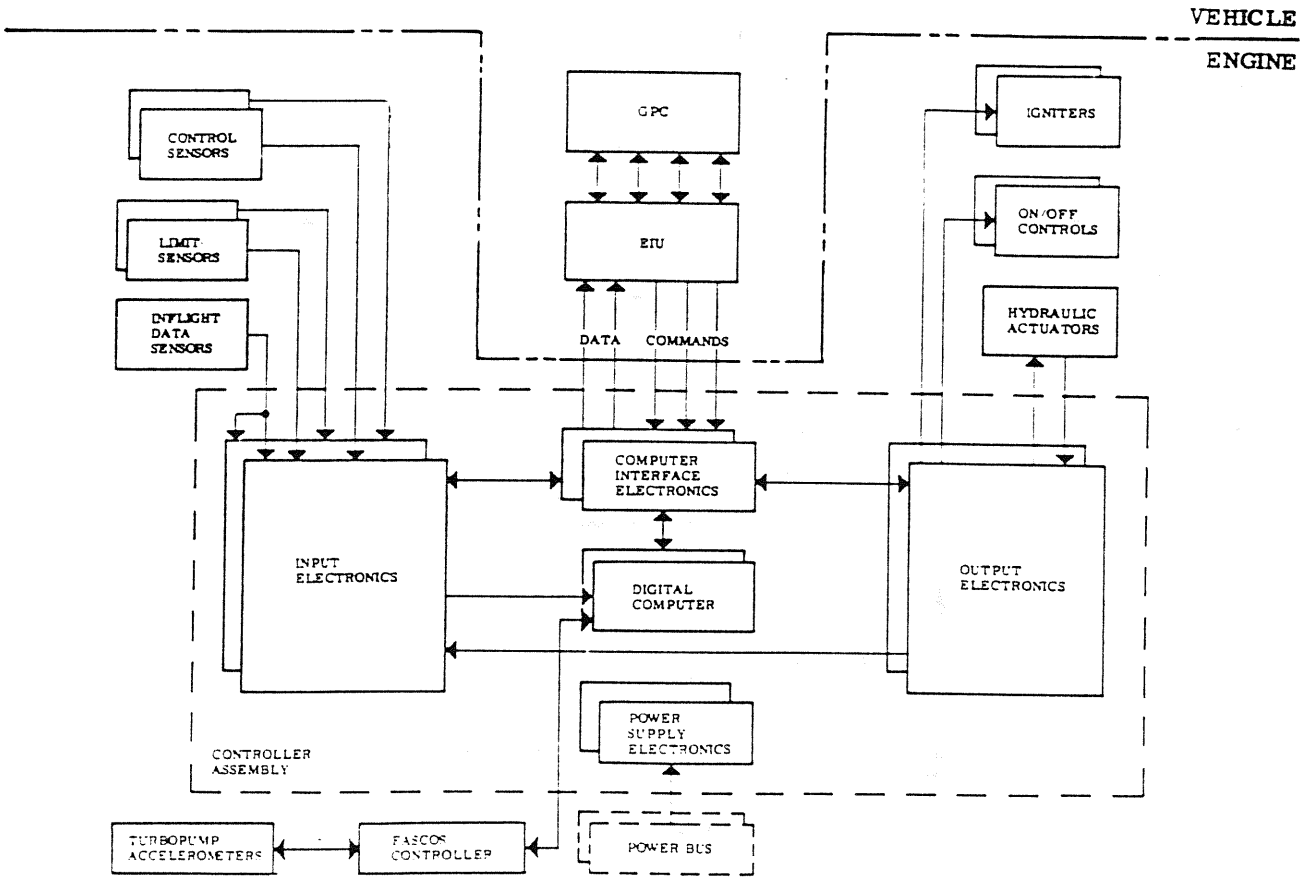


Figure 1.14-6.- Controller organization.

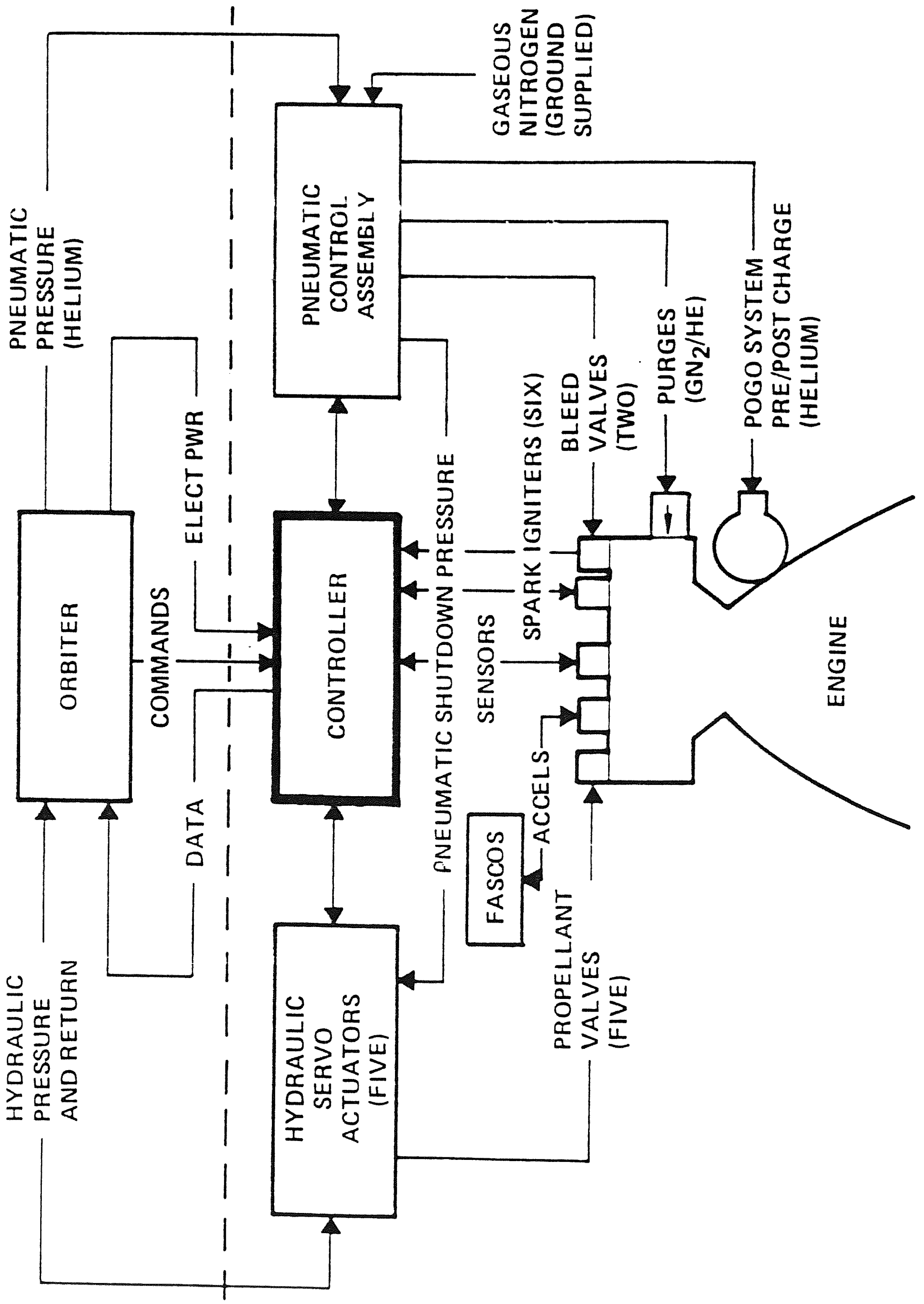


Figure 1.14-7.- SSME controller functions.

1.14.2.3 Computer Interface Electronics

The primary purpose of the CIE is to provide control of communication between all units within the controller, and to control the flow of data within the controller. The CIE controls input data to the DCU, and DCU output commands to the OE. The CIE provides the controller interface to the orbiter vehicle through the engine interface unit (EIU). Vehicle commands and requests for data are received through the vehicle/engine electronics interface (VEEI). The controller receives engine commands from the vehicle over triple-redundant channels. The controller transmits engine status and data to the vehicle over dual-redundant channels. Reference systems brief 1.13. The CIE also includes watchdog timers whose function is to determine which part of the dual-redundant system is in control. Reference SSSH drawing 10.5.

The CIE consists of the vehicle interface electronics and two identical computer input/output interface electronics channels. Each of the dual input/output interface electronics channels is dedicated to the associated computer in the DCU. In addition, each channel interfaces with the VEEI, IE, OE, and the power supply electronics. Power to the computer input/output interface electronics channels is supplied by the respective channels in the power supply electronics (PSE).

Both electrical output channels are isolated from each other, except for the following:

> The watchdog timer logic outputs are cross-fed to indicate to each computer the operational status of the other computer.
>
> The DMA write request and the 8-bit DMA address word is sent from the controlling computer to the standby computer via the IE. This allows the standby computer to update its own raw data base.

1.14.2.4 Digital Computer Unit

The controller consists of two identical general purpose digital computers which are independent of each other. Each DCU provides the computational capability necessary for all engine control functions. Each DCU receives sensor data, vehicle commands, and vehicle data request through its dedicated CIE channel. Each DCU performs computations necessary for full-authority closed-loop control of engine thrust and mixture ratio, and each DCU stores data until requested by the vehicle.

Normally, one DCU is in control and issues commands while the other DCU is in operational standby. Each DCU tests all control system components once per major cycle. The operation of the IE, OE, PSE, and dedicated CIE are continuously monitored. Thus, each computer has knowledge of the operational status of the controller and engine. Upon detection of an engine or controller failure, automatic corrective action is initiated by the controlling DCU. When in control, either DCU is capable of issuing commands to the engine control elements through either OE channel.

Each DCU consists of a central processor and a 2-mil plated-wire memory that uses nondestructive readout (NDRO) to store up to 16,384 17-bit (one bit for parity) data and instruction words. Typical computer instruction times are 2 microseconds for add and 9 microseonds for multiply. The total engine major cycle time is 20 microseconds (50 Hz).

1.14.2.5 Power Supply Electronics

The primary purpose of the PSE is to convert the 115 ± 5 V, three-phase, 400-Hz vehicle input ac power into the individual dc power supply voltage levels required by the controller subsystems. The controller receives ac power from two vehicle ac sources. The PSE consists of two identical channels each capable of deriving full power from one of the ac input power buses. Each channel supplies power to one channel of IE, OE, DCU, and CIE. Operating voltage for CIE channel C vehicle interface electronics is derived from the other two PSE channels. The PSE monitors the input ac power and the power supply operation. The PSE will initiate controller shutdown if the voltages are not within satisfactory limits. This is implemented by the PSE issuing a power failure sense (PFS) signal to the DCU if a failure of both input power busses or switching regulators occurs. Reference systems brief 1.13.

The PSE also contain the temperature control electronics, which is a nonredundant control circuit that maintains the controller internal temperature above -65° F by means of an internal heater. The temperature control electronics interface with vehicle dc power. These control circuits are electrically isolated from all other electronics within the controller.

This switch is located on panel R4. For all STS flights the heaters are not powered and the switch is left in the "off" position.

1.14.3 Fail-Operational/Fail-safe Design Features

Normal controller operation is under DCU A control. This DCU receives inputs from both IE A & B, and sends commands through both OE A & B. DCU B is operating and tracking DCU A. If DCU A fails, DCU B takes control, and would still receive inputs from IE A & B, and send commands through OE A & B. This provides fully operational avionics after the first failure. Engine control is transferred to the standby computer without impairing engine operation. If DCU B were to subsequently fail, the engine would perform a pneumatic shutdown sequence, ensuring a safe engine shutdown after the second failure. See figure 1.14-8.

During operation, the IE and OE are tested by the DCU and will be disqualified by the DCU if failed. A safe pneumatic shutdown will occur if both IE or OE fail. The power supplies are dedicated to controller channels, that is PSE A supplies DCU A, IE A, and OE A, PSE B supplies DCU B, IE B, and OE B. A safe pneumatic shutdown will occur if both PSE's fail.

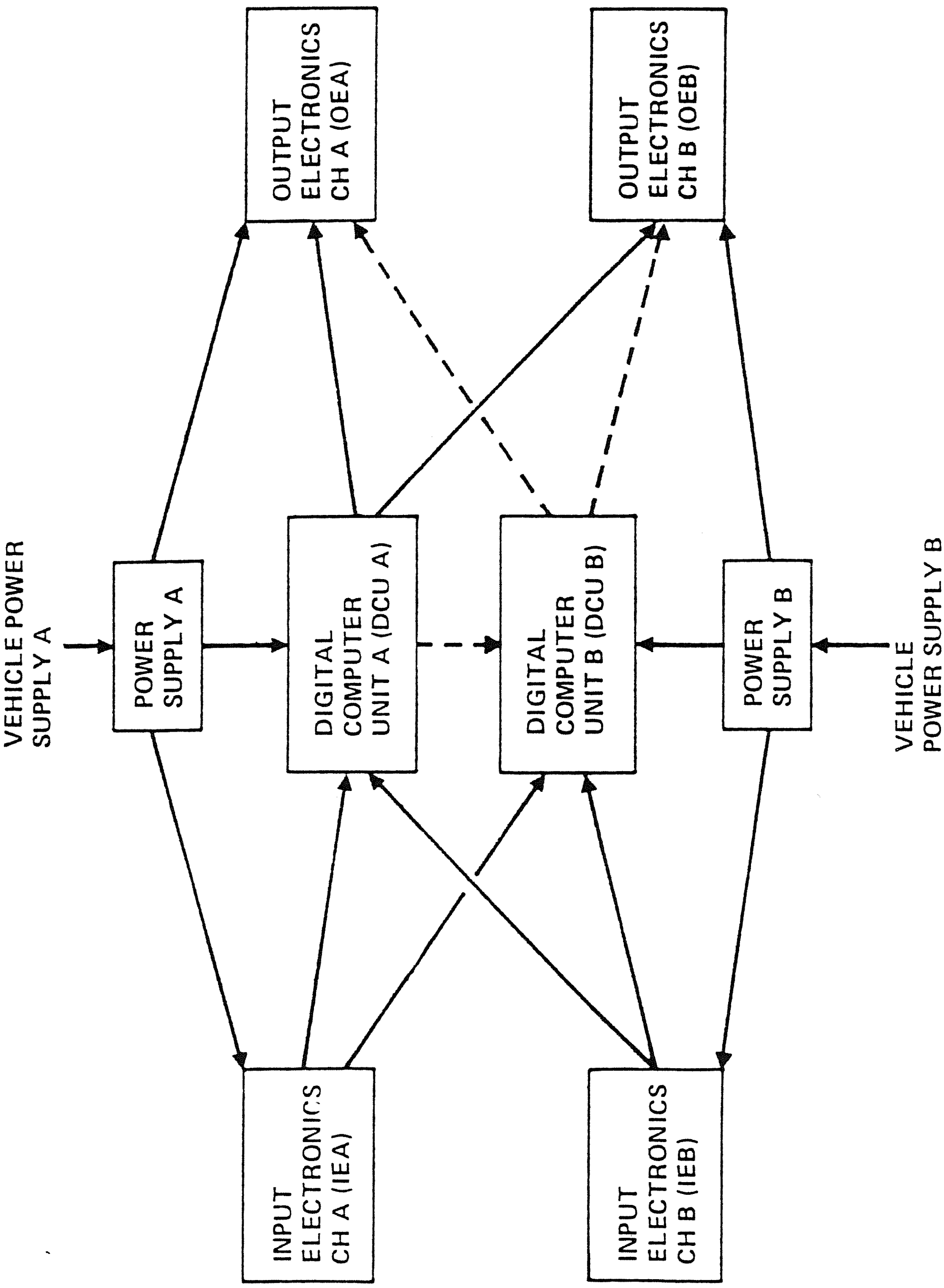


Figure 1.14-8.- SSME controller redundancy.

1.14.4 References

1. SSME Operation, Rocketdyne course no. ME-110(A) RIR

2. SSME Description & Operation, Rocketdyne no. RSS-8559-1-1-1

1.15 SSME "STARTBOX" VIOLATION TURNAROUND REQUIREMENT

This systems brief outlines the minimum turnaround time for engine startbox violation.

1.15.1 LO_2 Background

a. Prior to drainback initiation

A maximum LO_2 inlet temperature limit value of -287.7° F is set to ensure proper propellant conditioning and bleed flow. If LO_2 engine inlet temperatures are high, there could be improper bleed flow. If the LO_2 engine inlet temperatures are above the limit of -287.7° F, the LO_2 will become warm and less dense. During engine startup, the engine would demand more LO_2 because the available LO_2 is less dense. This could result in a catastrophic scenario because the turbine temps would build too rapidly causing the turbines to overspin.

The LO_2 in the "downcomer" is warmer and less dense than that in the external tank (ET) because a greater surface area is exposed to ambient temperatures and the bulk temperature of the tank does not change a great deal over time.

b. Subsequent to drainback initiation

LO_2 stable replenish is terminated at drainback initiation. The liquid oxygen then begins to travel from the tank, through the downcomer, the main engines, and out of the orbiter through the overboard bleed. As the warm LO_2 from the downcomer enters the main engines the inlet temperatures increase such that they enter the starting temperature regime of the engines - the startbox. Refer to ICD-13M15000 for Oxidizer Prestart Propellant Conditions - the startbox.

The engine startbox is violated when the LO_2 engine inlet temperatures fall below -289.2° F. Violating this temperature prior to T-31 seconds would cause the LO_2 to become too cold and dense. Because of the increased density of the LO_2, there may be an increase in the mixture ratio which could result in a catastrophic scenario because the turbine temps would build too rapidly and cause the turbines to overspin.

1.15.2 LH_2 Background

a. Prior to LH_2 tank prepress

Sixty minutes of continuous recirculation is required per OMRSD File 2 for engine temperature stabilization. This prevents the formation of bubbles in the fuel lines that could cause SSME fuel pump cavitation at ignition.

1.15.3 LO_2 Discussion

a. Subsequent to a launch hold for startbox violation

Thirty minutes are required to establish the correct valve configuration, to revert back to LO_2 stable replenish, and to reestablish LO_2 engine inlet temperatures.

An additional **30 minutes** is then required to maintain engine inlet temperatures within limits (inlet temps < -287.7° F) prior to drainback initiation. Proper thermal conditioning ensures that the engine is properly warmed to establish startbox conditions at liftoff. The 30-minute requirement is based on engine start data.

b. Drainback hold capability - refer to figure 1.15-1

With 30 minutes of stable replenish, approximately 2 minutes of hold capacity exists prior to startbox violation. The warm LO_2 from the facility does not have enough time to transgress up the full length of the downcomer during this 30-minute time frame. Therefore, there is a thermal barrier where the cold LO_2 from the tank meets the warmer facility LO_2. This thermal barrier is located somewhere downstream of the downcomer interface to the ET.

With 75 minutes of stable replenish, approximately 4 to 5 minutes of hold capability can be established. The warm LO_2 has time to transgress to the interface of the ET. Therefore, the thermal barrier is now located at the interface to the ET which provides extended hold capability through the additional amount of warm LO_2 in the downcomer.

1.15.4 LH_2 Discussion

Recirculation pump termination occurs at T-9.4 seconds. There is no impact to LH_2 system if a scrub occurs prior to this time because the hydrogen burn igniters - the "sparklers" - are ignited at T-10 seconds to burn any random hydrogen atoms that were not combined during the SSME combustion process. Replacement of these NSI's requires a few days to support a subsequent launch attempt.

1.15.5 Resuming Countdown

The countdown clock recycles to T-20 minutes for a revert due to engine startbox violation. Therefore, backing the T-20 clock off of the liftoff time and working this in conjunction with the two 30-minute holds, the countdown clock can be resumed 5 minutes subsequent to the start of stable replenish. The clock can also be resumed any time before this and hold at T-5 until 30 minutes of stable replenish is established. This timeframe is better illustrated in figure 1.15-2.

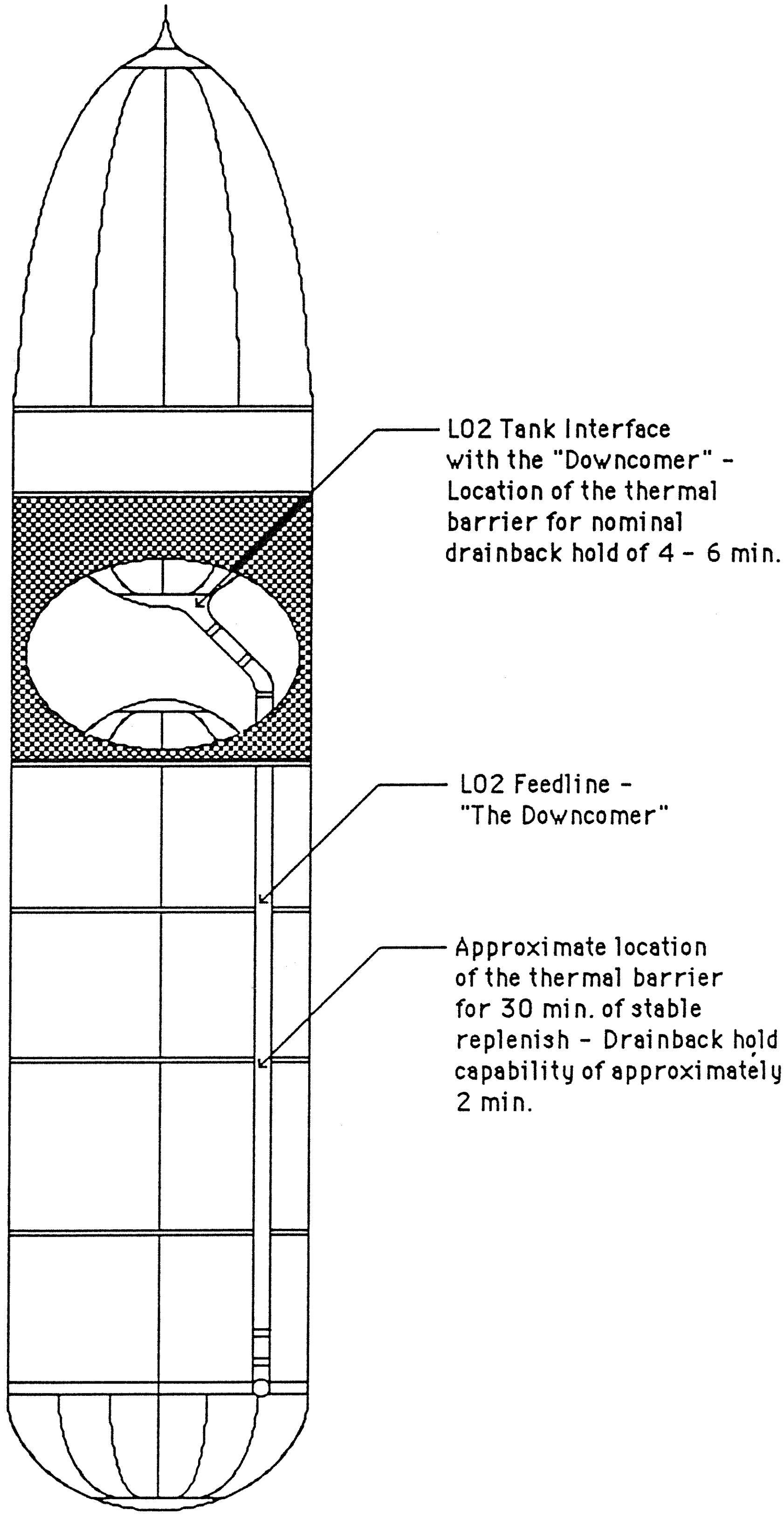


Figure 1.15-1.- Thermal barrier location.

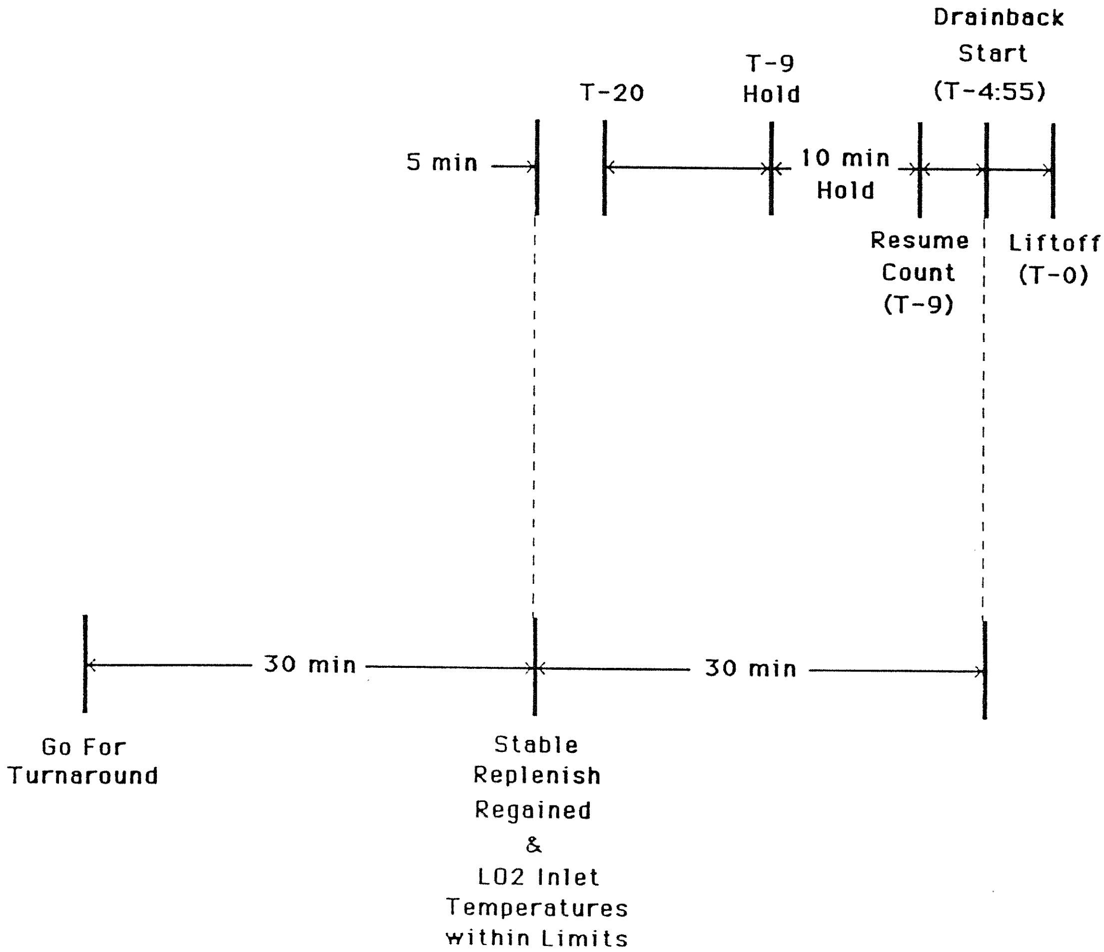


Figure 1.15-2.- Countdown timetable.

1.15.6 Conclusion

The minimum turnaround time from "go for turnaround" until drainback initiation is at least **60 minutes.** From "go for turnaround" until T-0 is at least **65 minutes.**

2.1 ET OVERVIEW

2.1.1 Introduction

The external tank (ET) serves as the structural backbone for the orbiter and the SRB's and contains the liquid hydrogen (LH_2) and liquid oxygen (LO_2) propellants for the three SSME's. The ET is 154 ft. long and has a diameter of 27.6 ft. It weighs approximately 69,000 lb empty and 1,660,000 lb when fully loaded with propellant at launch. Six lb of LO_2 are loaded for each lb of LH_2. Because the ET is expendable, the ET active components were kept to a minimum. The only active components are the vent/relief valves. All operational instrumentation is hardwired to the orbiter. All power and pressure are received from the Orbiter or ground facility.

This systems brief describes the primary structures, the propulsion system, and some of the instrumentation.

2.1.2 Structural Components

The ET has three primary structures: an LO_2 tank, an intertank, and an LH_2 tank. These structures are shown in figure 2.1-1. Both tanks are constructed of aluminum alloy skins with support or stability frames as required. The intertank aluminum structure utilizes skin stringers with stabilizing frames. The primary aluminum materials used for all three structures are 2024, 2219, and 7075 alloys. Low density foam insulation and high density composite material are used as thermal insulation for the cryogenic tanks and to protect against ascent heating rates. There are seven major types used, each with their own heat transfer rates, bondline temperature limits, and structural strength.

2.1.2.1 LO_2 Tank

The LO_2 tank, shown in figure 2.1-2, is located at the top of the ET and has an ogive shape to reduce aerodynamic drag and aerothermodynamic heating. The ogive nose section is capped by a flat removable cover plate and a nose cone. The nose cone consists of a removable conical assembly that serves as an aerodynamic fairing for the propulsion and electrical system components. The forward most element of the nose cone functions as a cast aluminum lightning rod. The LO_2 tank volume is 19,672 ft^3 at 22 psig and -297° F (cryogenic).

All loads except aerodynamic loads are transferred from the LO_2 tank at a bolted, flange-joint interface with the intertank.

The LO_2 tank also includes an internal slosh baffle and a vortex baffle to dampen fluid slosh. The vortex baffle is mounted over the LO_2 feed outlet to reduce fluid swirl resulting from slosh and to prevent entrapment of gases in the delivered LO_2.

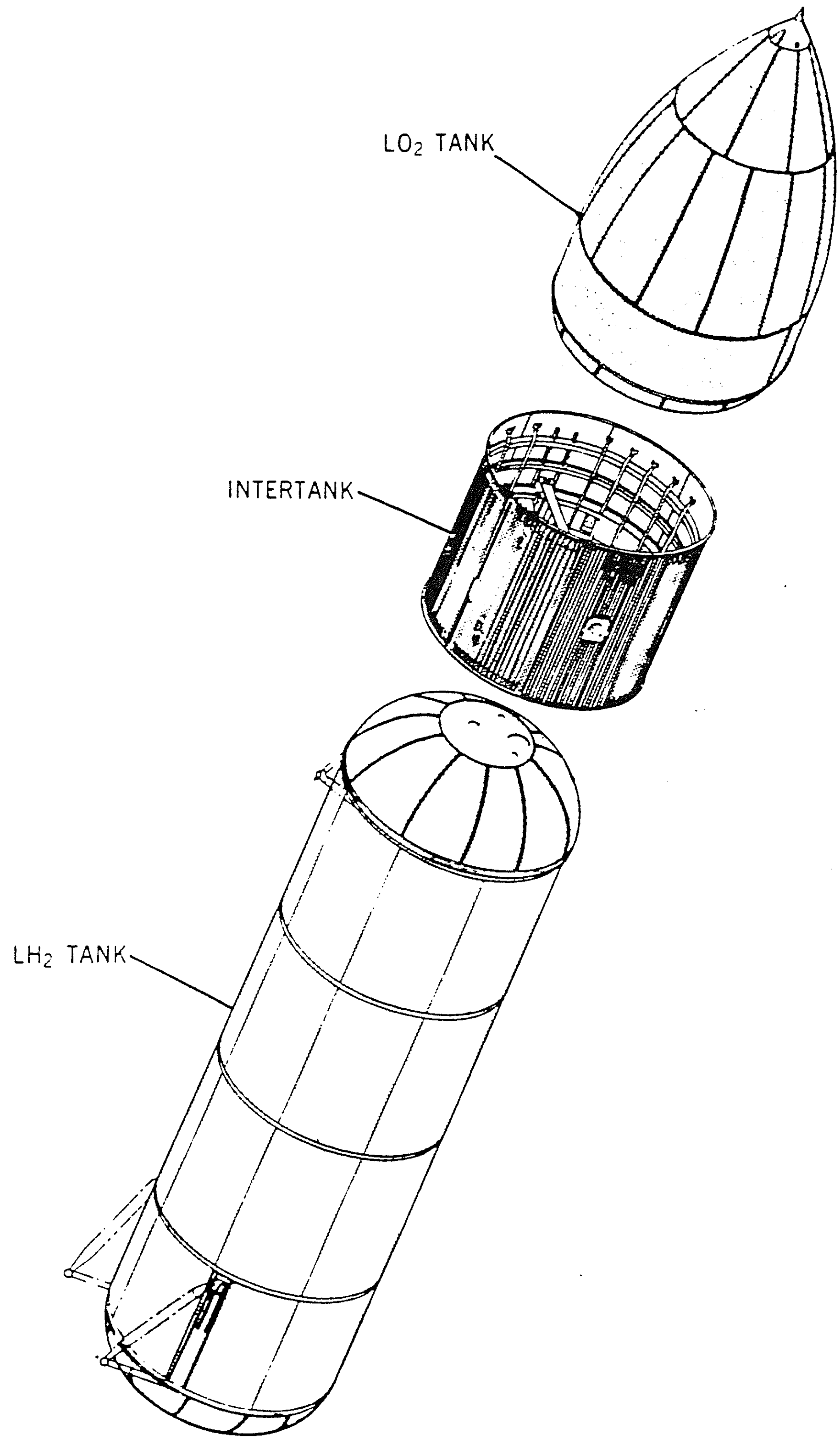
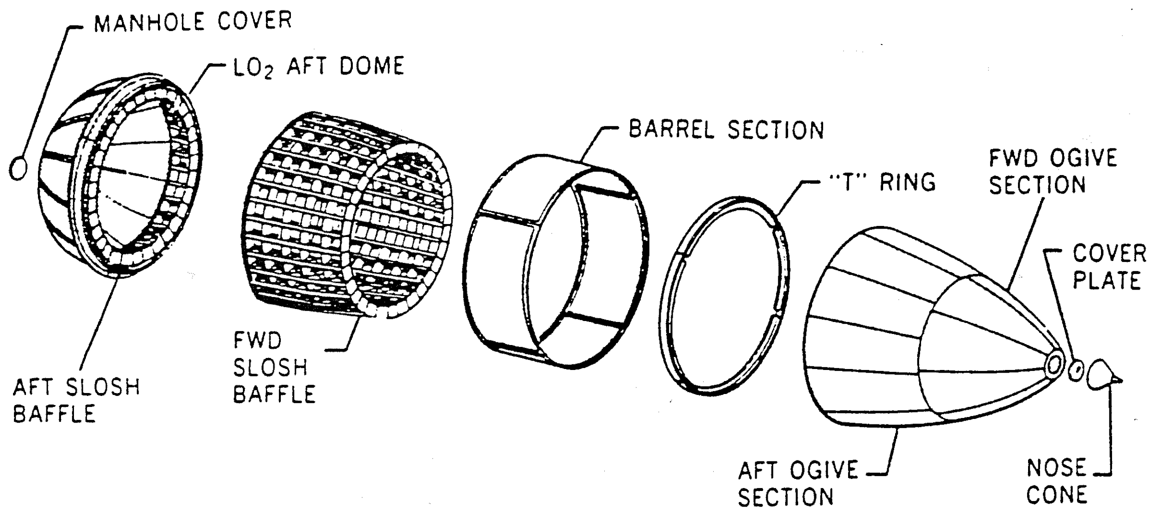
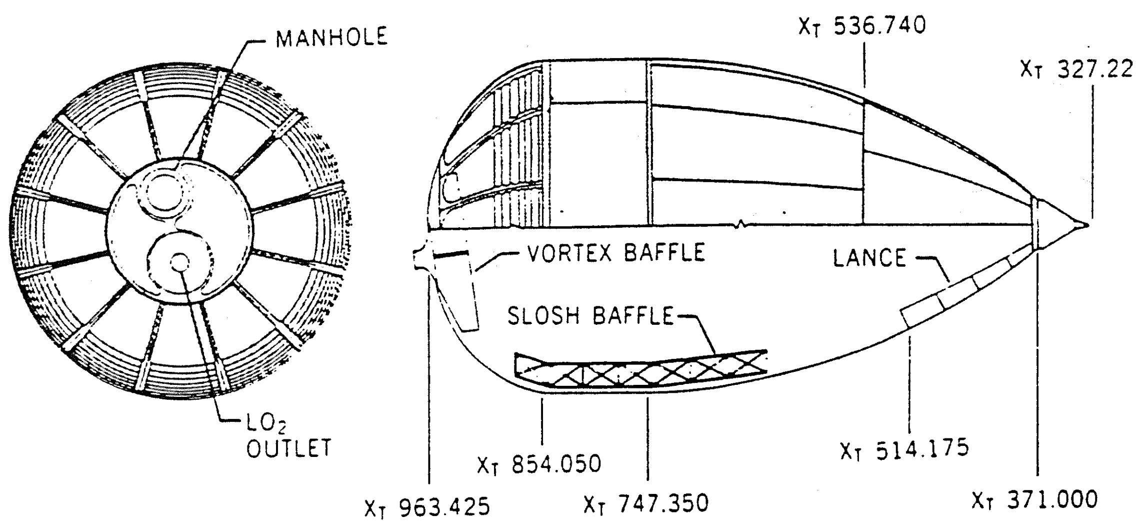


Figure 2.1-1.- External tank structure.



LO_2 TANK STRUCTURE



LO_2 TANK PROFILE

Figure 2.1-2.- LO2 tank.

2.1.2.2 Intertank

The intertank, shown figure 2.1-3, is the ET structural connection which joins with both the LO_2 and LH_2 tanks. Its primary functions are to receive and distribute all thrust loads from the SRB's and transfer loads between the tanks.

The SRB two forward attach fittings are located 180° apart on the intertank structure. A beam is extended across the intertank structure and is mechanically fastened to the attach fittings. When the SRB's are firing, the beam will deflect due to high stress loads. These loads will be transferred to the fittings.

Adjoining the SRB attach fittings is a major ring frame. The loads are transferred from the fittings to the major ring frame which then distributes the tangential loads to the intertank skin. Two panels of the intertank skin, called the thrust panels, distribute the concentrated axial SRB thrust loads to the LO_2 and LH_2 tanks and to adjacent intertank skin panels. These adjacent panels are comprised of six stringer-stiffened panels.

The intertank also functions as a protective compartment for housing the operational instrumentation and range safety components.

2.1.2.3 LH_2 Tank

The LH_2 tank, as shown in figure 2.1-4, is the bottom portion of the ET. The tank is constructed of four cylindrical barrel sections, a forward dome, and an aft dome. The barrel sections are joined together by five major ring frames. These ring frames receive and distribute loads. The forward dome-to-barrel frame distributes the loads applied through the intertank structure is also the flange for attaching the LH_2 tank to the intertank. The aft major ring receives orbiter-induced loads from the aft orbiter support struts and SRB-induced loads from the aft SRB support struts. The remaining three ring frames distribute orbiter thrust loads and LO_2 feedline support loads. Loads from the frames are then distributed through the barrel skin panels. The LH_2 tank has a volume of 53,166 ft^3 at 29.3 psig and -423° F (cryogenic).

The forward and aft domes have the same modified ellipsoidal shape. For the forward dome, mounting provisions are incorporated for the LH_2 vent valve, the LH_2 pressurization line fitting, and the electrical feed-through fitting. The aft dome has a manhole fitting for access to the LH_2 feedline screen and a support fitting for the LH_2 feedline.

The LH_2 tank also has a vortex baffle to reduce swirl resulting from slosh and to prevent entrapment of gases in the delivered LH_2. The baffle is located at the siphon outlet just above the aft dome of the LH_2 tank.

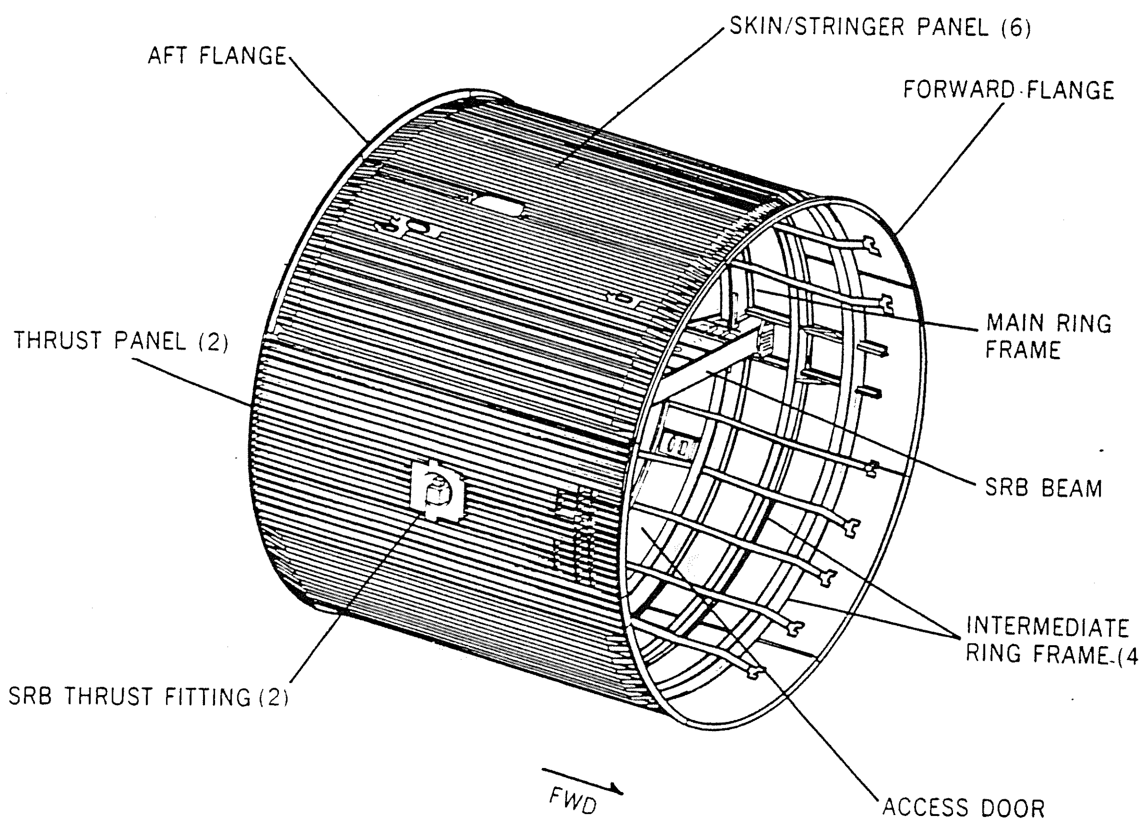
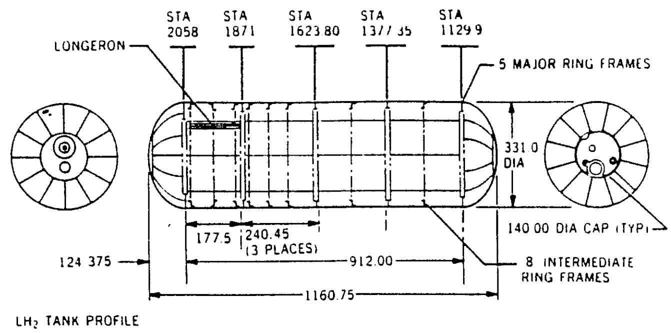


Figure 2.1-3.- Intertank structure.



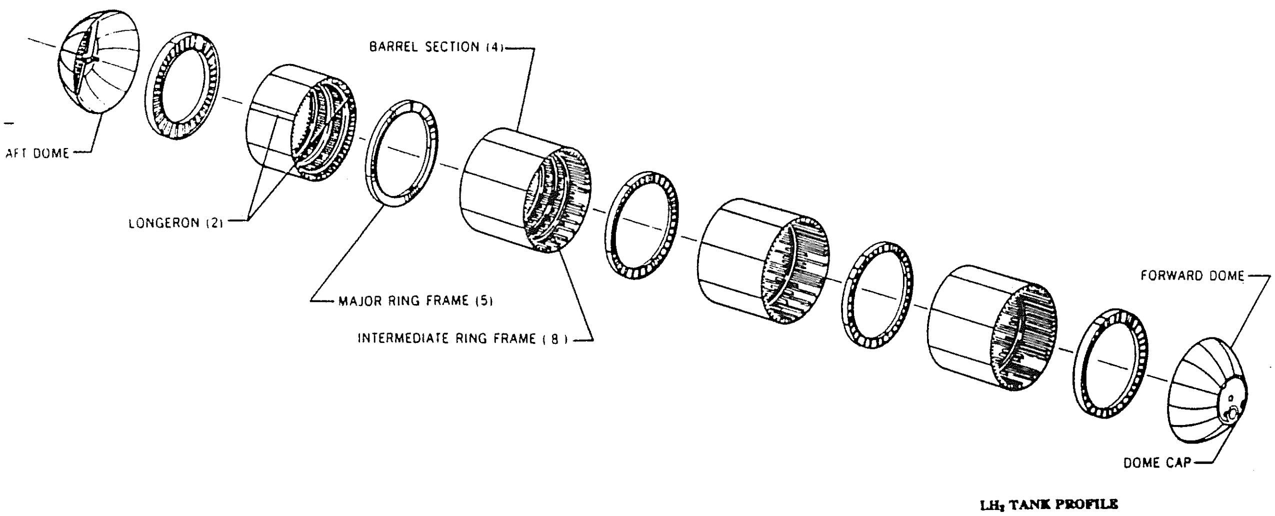


Figure 2.1-4.- LH_2 tank.

2.1.2.4 ET Lines

Over a dozen major flow lines exist in the ET. Their functions include: propellant transfer to the main engines, LO2 and LH2 tank pressurization, cryogenic boiloff venting, hazardous gas detection systems (HGDS), and purges. A listing of these major lines and mechanical joints is given in Table 2.1-I. Figure 2.1-5 also shows the primary ET lines.

2.1.3 Propulsion System

The prime function of the ET is to deliver the oxidizer and fuel to the orbiter for the SSME's. To deliver the propellants requires a LO_2 system, a LH_2 system, a LO_2 and a LH_2 tank pressurization system, and a LO_2 and LH_2 vent/relief system. The primary ET flow lines are shown on figure 2.1-5.

2.1.3.1 LO_2 Feed System

The LO_2 feed system, shown in figure 2.1-6, consists of the 17-inch insulated LO_2 feedline and the helium inject line. The feedline is attached to the outlet of the aft dome of the LO_2 tank and is routed through a port in the intertank skin. From there, the line runs down the side of the LH_2 tank to the LO_2 disconnect on the right ET/orbiter umbilical-disconnect plate.

The helium inject system introduces a controlled flow of gaseous helium from the launch facility through a 3/8-inch line into the aft end of the LO_2 feedline to prevent geysers during propellant loading and holds before launch. A geyser can occur in vertical systems with a tank and a long feedline for cryogenic oxygen. Heat transfer into the line causes GO_2 bubbles to form and begin rising in an LO_2 line. As the bubbles rise, they coalesce to form bigger bubbles. In the feedline, the result is an expanding vapor pocket as wide as the feedline which expels the liquid above it into the tank with damaging force. The expanding bubble also lowers the head pressure on the LO_2 below. When liquid then reenters the line, a large water hammer results. The purpose of the helium inject is to provide evaporative cooling of LO_2 in the feedline (LO_2 evaporates into the helium bubbles) to reduce the possibility of LO_2 reaching local saturation temperature which could cause a geyser.

2.1.3.2 LH_2 Feed System

The LH_2 feed system, shown in figure 2.1-7, consists of a 17-inch feedline and a 4-inch recirculation line. The feedline is attached to a bell-mouth within the tank near the bottom of the aft dome. From there, the line runs through a flanged port on the upper LH_2 tank aft dome to the LH_2 disconnect on the left ET/orbiter umbilical disconnect plate. During engine preconditioning in the launch countdown, the recirculation line carries warm LH_2 from the engine back to the LH_2 tank. Electrically driven recirculation pumps deliver the flow.

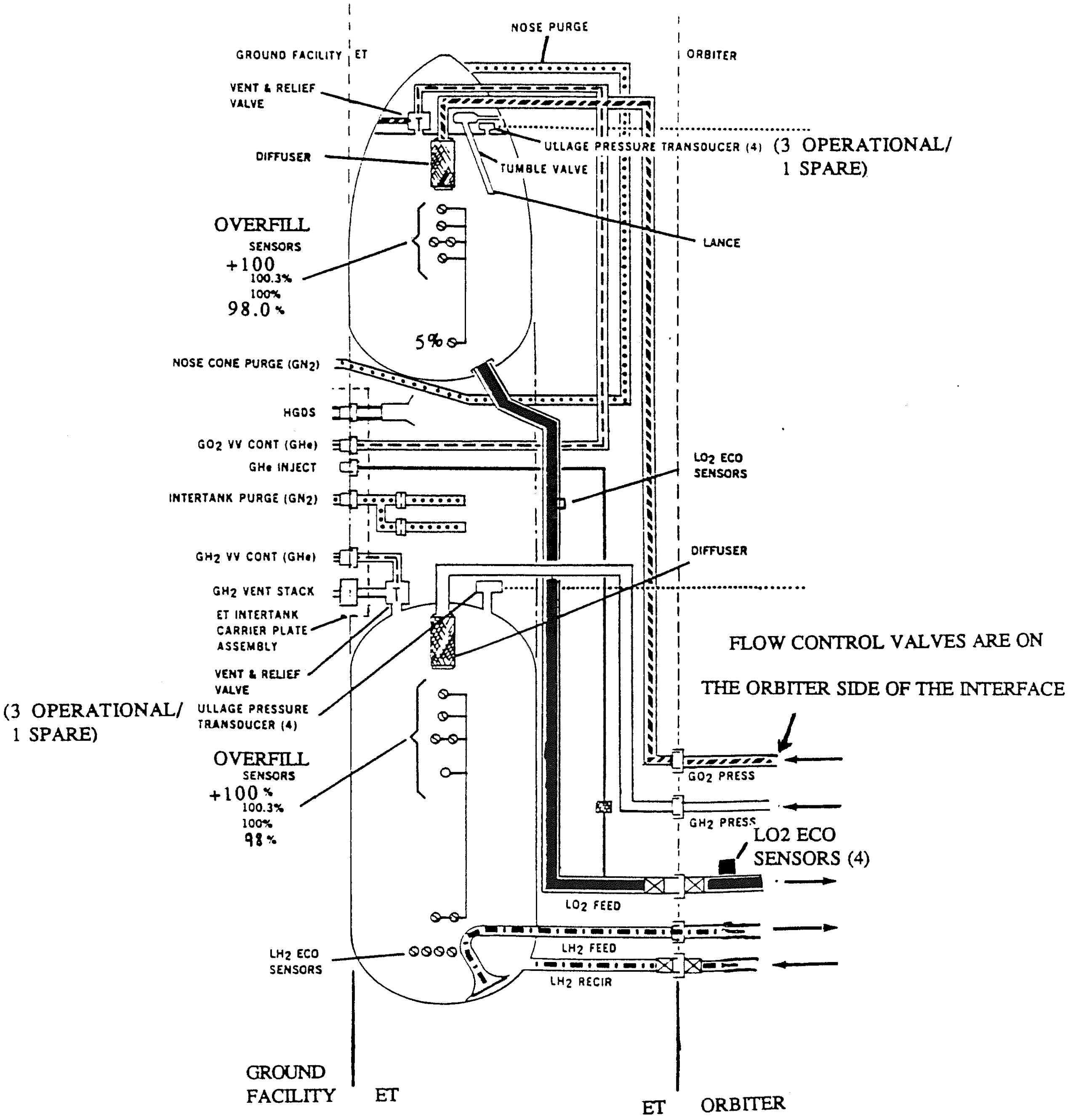


Figure 2.1-5.- External tank lines.

TABLE 2.1-I.- ET LINES AND JOINTS

Line description	Line size (inches)	Material	No. of sect	Mechanical joints			Flexible joints				Insulation
				No.	Type	Seal	Bellows	Gimbal/ bellows	Univ	Pin	
LO_2 feed	17	CRES and aluminum	9	9	Bolted flange	Raco/ Creavey*	--	5	--	--	CPR-488
LH_2 feed	17	CRES and aluminum	4	4	Bolted flange	Raco/ Creavey*	4	--	--	--	SLA-561/CPR-488/ Cryopumped jacket (ext. sect)
LH_2 recircu- lation	4	CRES	1	2	Bolted flange	Raco/ Creavey*	2	--	--	--	SLA-561/CPR-488 Cryopumped jacket around bellows
Helium inject	3/8	CRES	27	50	MS and B-nut						
GO_2 press	2	CRES	9	10	Bolted flange	Naflex	1	8			
GH_2 press	2	CRES	8	9	Bolted flange	Naflex	--	11			
GO_2 vent	5 1/8	CRES	3	3	V-band	Raco	2				
GO_2 vent actuation	1/4	CRES	10	20	MS and B-nut	K					
GH_2 vent	5	CRES	1	2	Bolted flange	Raco/ Creavey	1	1	--	--	BX-250
GH_2 vent actuation	1/4	CRES	2	5	MS and B-nut	K					
Intertank purge	3/8 & 1/2	CRES	1	2	MS and B-nut						
	2	Aluminum	6	4	V-band coupl.						
Nosecap purge	3/8 & 1/2	CRES	--	--	MS and B-nut	K					
HGDS**	1/4	CRES	4	3	MS and B-nut						

*Naflex seal at disconnect.

**Hazardous gas detection system.

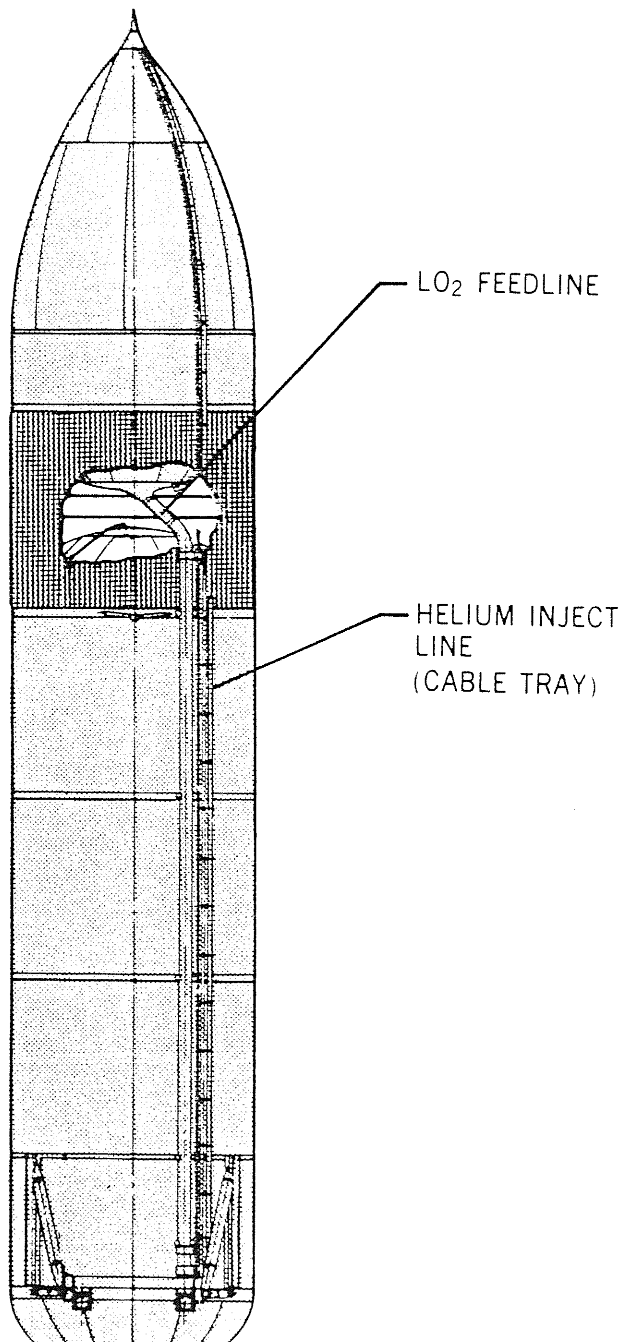


Figure 2.1-6.- LO_2 feed system.

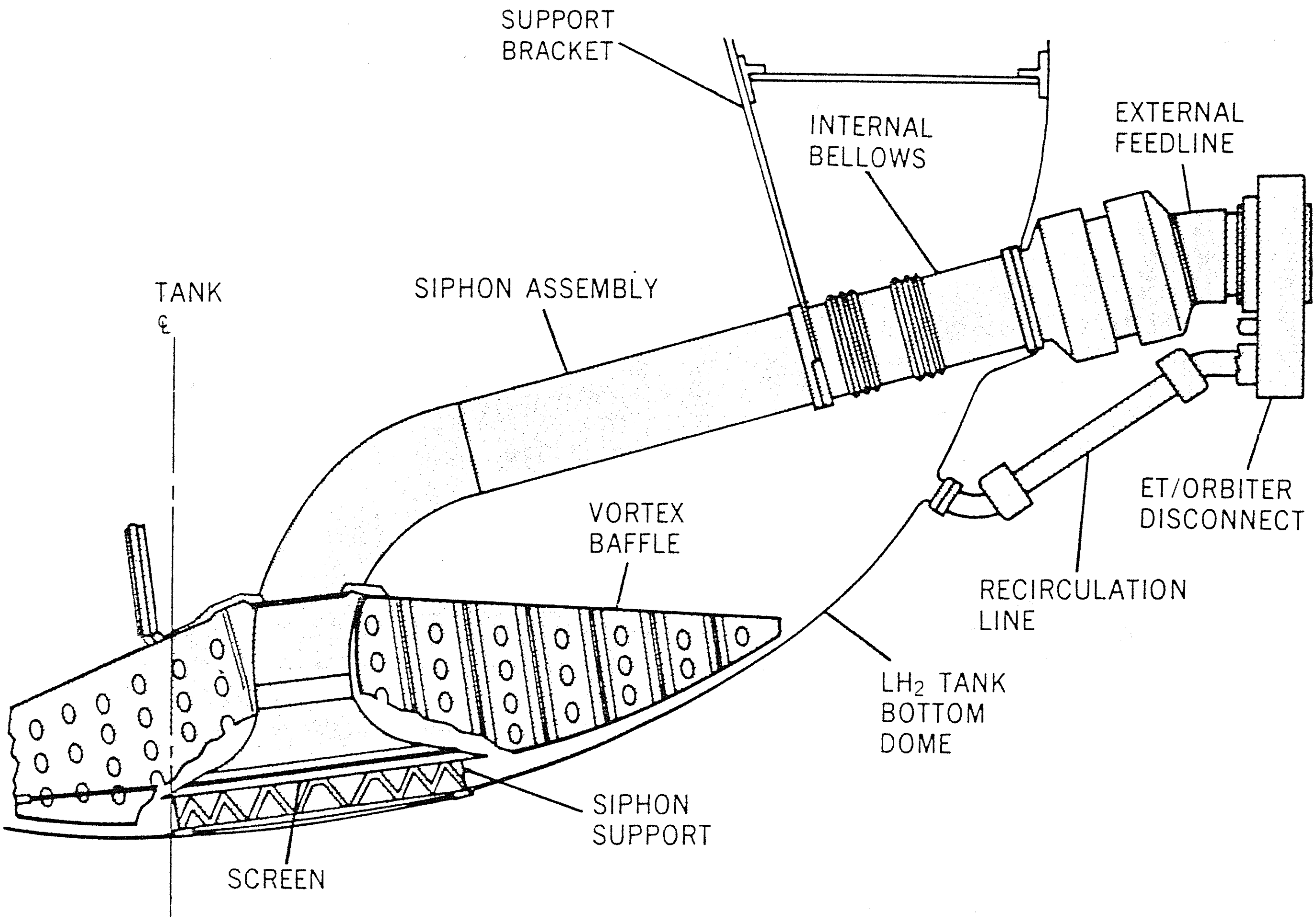


Figure 2.1-7.- LH_2 feedline.

2.1.3.3 LO_2/LH_2 Tank Pressurization Subsystems

The LO_2 and the LH_2 tanks are pressurized by gaseous propellant routed back to the tanks from the SSME's. Both the GO_2 and GH_2 pressurization lines are 2-inch lines. The GO_2 pressurization line extends from the LO_2 tank to the right aft ET/orbiter umbilical disconnect plate; the GH_2 pressurization line extends from the LH_2 tank to the left umbilical disconnect plate. To transfer the propellants to the engines, a turbopump feed system is used. To pressurize the LO_2 and LH_2 tanks, gaseous oxygen from the LOX heat exchanger and gaseous hydrogen tapped off from the low pressure fuel turbopump turbine exhaust are routed back to the ET. The desired tank ullage pressures are maintained by the operation of three flow control valves for each propellant system. The GH_2 system uses active flow control valves. The GO_2 system uses flow control valves shimed to a fixed orifice configuration.

2.1.3.4 Propellant Tank Vent/Relief

The ET has the capability to vent excess LH_2 and LO_2 that was converted into gases during the ET propellant loading operation. The ET also has the capability to relieve automatically any pressure buildup within the LH_2 and LO_2 tanks over predetermined limits. If the ET vent/relief valves fail to work properly, the crew will receive an alert message once 46.0 psia has been exceeded for the LH_2 tank and 29.0 psig has been exceeded for the LO_2 tank. Note that the LH_2 pressure is absolute while the LO_2 is gauge pressure.

2.1.4 Instrumentation

The primary instrumentation includes the ullage pressure sensors, the liquid level sensors, and the low level sensors (ref. fig. 2.1-5). All instrumentation and power to the ET and SRB's are channeled through one of two pathways. These are the ET/orbiter LO_2 monoball and the LH_2 monoball. These monoballs are adjacent to the LO_2 and LH_2 12-inch feedlines at the ET/orbiter interface.

2.1.4.1 Ullage Pressure Sensors

The LO_2 and LH_2 tanks each have four ullage pressure sensors. For flight, three ullage pressure sensors are used to control the ullage pressure. The fourth sensor is a spare that can be switched in prelaunch to replace a failed sensor. The LH_2 sensor has a pressure range of 12 to 52 psia. The system error for the LH_2 tank pressure range of 32 to 34 psia is ±0.4 psi, and the system error throughout the remaining pressure range is ±1.2 psi. The LO_2 sensor range is 0 to 30 psig. The system error for the LO_2 tank pressure is ±0.3 psi for tank pressure range of 20 to 22 psig and ±0.9 psi for the remaining pressure range. The sensors are potentiometer-type, absolute-pressure transducers with a voltage output of 0 to 5 V dc directly proportional to the amplitude of the absolute pressure present.

To maintain the desired ullage pressure, the flow control valves are automatically opened if the tank pressure drops out of the control band. The LH_2 flow control valves can be manually opened by the crew if necessary. There is no manual control for the LO_2 flow control valves. The LO_2 flow control valves have been replaced with fixed orifices. This results in a wider pressure band profile than previously existed, but at the same time is a more reliable LO_2 ullage pressure control system.

2.1.4.2 Liquid Level Fill Sensors

Liquid level sensors are used in both propellant tanks to indicate the presence (wet indication) or absence (dry indication) of LO_2 or LH_2 in the tanks. The sensors are platinum wires that change resistance when wetted by cryogenic propellants. The fill sensors are located at 5 percent (one sensor), 98 percent (two sensors), -100 percent (one sensor), 100 percent (two sensors), +100 percent (one sensor), and overfill (one sensor) levels of each LO_2 and LH_2 tank (fig. 2.1-5). All these indications are monitored during prelaunch fill operations. The 5 percent level marks are cues indicating propellant depletion will occur in 25 to 30 seconds during a normal ascent. The 5 percent sensors are tripped to the dry state when 75,651 lb LO_2 and 11,899 lb LH_2 are left in the MPS system (ET and orbiter). This reflects 70,850 lb LO_2 and 11,534 lb LH_2 in the ET only.

2.1.4.3 Low Level Sensors (ECO sensors)

There are four LO_2 and four LH_2 liquid level sensors that are used to monitor for propellant depletion. The LO_2 sensors are located in the orbiter feedline; the LH_2 sensors are located in the LH_2 tank bottom. These sensors are platinum wires like the liquid level sensors that are used for tanking. During a nominal mission they will not get tripped to the dry mode, as nominal MECO will occur once a velocity cue is reached, leaving enough residual LO_2 and LH_2 to cover the engine cutoff (ECO) sensors.

If not enough LO_2 or LH_2 exists to reach nominal MECO velocity, the ECO sensors will go dry and command an early MECO to prevent propellant depletion. Propellant depletion causes severe SSME damage. Early MECO will cause an underspeed at MECO, requiring more OMS propellant to reach orbital velocity targets.

The GPC's will not recognize the ECO sensors going dry to command an early MECO unless an ARM command has been sent previously. The ARM command is sent at the arming mass, nominally about 10 seconds before expected MECO velocity (about 8 minutes 20 seconds). An ARM command will also be sent any time two SSME's are shut down before MECO.

2.2 ET PROPELLANT LOADING

2.2.1 General Loading Requirements

The external tank (ET) is loaded with the cryogenic propellants, liquid hydrogen (LH_2) and liquid oxygen (LO_2) in the final countdown phase before launch. The loading is controlled automatically by the Launch Processing System (LPS) with operators available to monitor and control any off-nominal operations. The loading operations have been designed to load the propellant necessary for ascent, to control the density and liquid level (quantity) of the propellants, and to control system parameters. Parameter control is required to meet structural and thermal protection system constraints, and to prevent LO_2 "geysering." The loading accuracies required are ±0.43 percent for LO_2 and ±0.35 percent for LH_2. It is also necessary to maintain an inert atmosphere in the intertank and disconnect cavities and to minimize ice/frost accumulation on areas that could cause damage to the orbiter tiles and prevent the accumulation of hazardous concentrations of flammable gas.

For the loading operation, the ET interfaces directly with the facility through a Ground Umbilical Carrier Plate (GUCP) that separates at lift-off through actuation of a pyrobolt or by secondary actuation of a lanyard system that fractures the bolt. The services supplied through the GUCP are electrical functions, helium actuation sources to open/close the vent valves, intertank gaseous nitrogen (GN_2) purge to maintain an inert atmosphere and temperature controlled environment, helium inject for the "anti-geyser" system, a sense line to collect intertank gases for a facility mass spectrometer used for evaluation of hazardous gas conditions in the intertank, and a heated GN_2 purge to control nose cap temperature and to maintain an inert atmosphere.

The LO_2 propellant feed system interfaces with the orbiter through a 17-inch quick disconnect to fill and drain the tank. This fill/drain operation is accomplished using the orbiter LO_2 system, which interfaces with the facility through a disconnect that separates at launch. During filling, the LO_2 flows from the facility through an 8-inch orbiter fill line, then through the orbiter manifold and 17-inch feedline, then up the long 17-inch ET feedline to the contoured no-drop-out tank outlet and into the tank. A horizontal geyser splash plate precludes a geyser from splashing up, super cooling the ullage gas and collapsing the tank. The tank outlet also contains an antivortex baffle and an 800-micron screen needed during engine operation.

On ET's 1 through 4, a 4-inch line connected the tank outlet to the aft portion of the LO_2 feedline to provide circulation during filling to avoid a geysering situation. All remaining ET's do not have this anti-geysering line but depend on direct helium injection into the feedline to prevent a geyser from forming. Geysering is discussed in greater detail in section 2.2.2.

The LH_2 is loaded through the LH_2 feed system, which is similar to the LO_2 feed system except that a unique siphon concept is utilized to provide maximum LH_2 usable propellant to the orbiter interface without a cumbersome external feed system that would be exposed to high heating rates from the SRB and main engine plumes. Vortex baffles are also employed, as well as a 400-micron screen in the siphon. The siphon contour is identical to the LO_2 outlet to minimize drop-out.

A 4-inch recirculation line connects the orbiter to the LH_2 tank to pre-condition the SSME's before engine start. Also, LH_2 from the facility flows through this line during the replenish period.

The normal time line for loading LO_2 and LH_2 and the times various vehicle valves are cycled are shown in figure 2.2-1. The 4-hour Shuttle System Spec. vol. X requirement can be met with this time line by reducing the replenish time to 1 hour, which is sufficient for stable replenish. Paragraph 3.2.1.2.2 of this document has been waived for Kennedy Space Center (KSC) to permit a 4-hour launch from standby. The original 2-hour requirement cannot be met at KSC without facility modifications.

Paragraph 3.2.1.2.3.2 of vol. X requires the capability to hold for 7 hours after loading and before start of LO_2 drainback. Also, a 2-minute hold capability is required after start of drainback to T0-25 seconds, without reduction of vehicle performance. These requirements can be met. The hold capability after start of drainback can be increased on the day of launch by assessing some of the parameters in the flight performance reserve (FPR); e.g., winds, and trading any reduction in FPR for available hold time.

2.2.2 LO_2 Anti-geysering System

On the first four ET's, a 4-inch line connected the tank outlet to the aft portion of the LO_2 feedline to provide LO_2 circulation during fill. This line is called the anti-geyser line. Helium was injected in the anti-geyser line to provide a jet pumping action as well as to reduce the density in the line. This caused cold LO_2 from the tank to flow down the main feedline during LO_2 loading at low flow rates because of the density difference between the two lines.

All following ET's do not have an anti-geyser line but depend on direct injection of helium into the bottom elbow of the ET 17-inch feedline to prevent a geyser from forming. The action of the helium as it bubbles through the LO_2 is to vaporize a portion of the propellant, thereby cooling the remainder. This type of geyser protection is called main feedline injection (MFI).

Six hundred and ninety pounds of weight reduction on the lightweight ET (LWT) is attributed to deletion of the anti-geyser line and the incorporation of a helium inject into the main feedline. A significant cost per flight savings ($113,000) was also realized. A reduction in margin to preclude a geyser during LO2 loading is a drawback of the change. However, loading procedures and controls have been instituted to maintain safe operation. Reduction in margin exists at low fill flow rates for cases where an on-pad abort would occur. Margin as a function of fill flow rate is described in figure 2.2-2. Exceeding limit lines would cause saturated LO_2 to form in the feed system, resulting in a geyser. After the geyser, which would empty portions of the feedline, a refill water hammer could cause damaging results. As seen in figure 2.2-2, at flow rates below 68 lb/sec, lower temperatures are required without the anti-geyser line; however, the limits are above the interface control document (ICD) requirements that the system is designed for. In case of an on-pad abort, LO_2 is stagnated in the orbiter feedlines because the prevalves are closed and the orbiter fill and drain valve is closed. Heat soak back from the engines could cause LO_2 in these lines to reach saturation in 12 minutes (minimum), resulting in a geyser. With the anti-geyser line installed, subcooled LO_2 is circulated down the ET feedline and circulates in the orbiter lines to maintain subcooled temperatures. For LWT, procedures have been incorporated to initiate LO_2 flow through the overboard bleed system within 2 minutes after on-pad abort or initiate helium bubbling. A secondary method is to provide a LO_2 drain. Operations without the anti-geyser line were verified on full-scale ground main propulsion system (MPS) testing, which simulated the ET, orbiter, and facility (although control was manual rather than with the LPS).

2.2.3 Facility Loading System

The locations of the LO_2 and LH_2 storage tanks with respect to the mobile launch platform are shown in the facility plan view in figure 2.2-3. Sketches of the LO_2 and LH_2 loading systems are found in figures 2.2-4 and 2.2-5, respectively, and schematics of the LO_2 and LH_2 systems are found in figures 2.2-6 and 2.2-7. The loading systems are controlled automatically by LPS consoles in the launch control center.

The LO_2 facility includes a 900,000-gallon storage tank and two pumps, either of which can deliver the required flow rate of LO_2. Also included in the storage facility are the valves and piping required to chill the pump, control flow at the various flow rates required, and provide for draining the ET after an FRF, a launch pad abort, or a tanking test. The vehicle has been designed to accept 5000 gal/min of LO_2 during loading, but the KSC system can deliver only about 1400 gal/min maximum. To increase this flow rate, a new 8-inch diameter vacuum-jacketed fill line is required, as well as installation of a high-capacity pump. The present fill line is a 6-inch vacuum-jacketed line about 1400 feet in length.

The cross-country fill line delivers the LO_2 to the Mobile Launch Platform (MLP), where there are instruments, a valve complex, and filters to control and filter the LO_2 entering the orbiter. The LO_2 is introduced into the orbiter through the LO_2 tail service mast (TSM) and the T-0 quick disconnect. A 10-inch vent line runs from the TSM to a dump basin to vent gaseous oxygen (GO_2) and unusable LO_2.

The LH_2 facility consists of an 850,000-gallon storage tank, and the valves and piping required to deliver LH_2 to the ET. The LH_2 is fed by pressure rather than pumping. This is accomplished by introducing some of the LH_2 to a vaporizer where it vaporizes and expands. This gas then flows to the top of the storage tank, where it pressurizes the ullage space in the tank. A 10-inch vacuum jacketed cross-country line delivers the LH_2 to the TSM and then to the ET in a manner similar to that used for LO_2.

A burn pond is provided in the LH_2 system to burn off any unusable hydrogen gas. A line leads from the TSM to the burn pond to dispose of unusable gas at this point. Gas from the high point bleed system also flows through this line. Another line goes to the burn pond from the GUCP to deliver the vented GH_2 from the ET for safe disposal.

The LH_2 system has been designed to deliver 12,000 gal/min of LH_2 during the fast-fill period. This was reduced to about 8400 gal/min during the early STS operations because of an ET back pressure requirement until the tank was full. Structural and thermal protection system (TPS) constraints resulted in this back pressure requirement. It is not presently known if this back pressure constraint will remain in the program.

A listing of the cyrogenic propellant properties of LO_2 and LH_2 is provided in table 2.2-I. The LO_2 and LH_2 flow rates, pressures, and temperatures at the orbiter inlet are shown in table 2.2-II for the various phases of propellant loading. This table shows the ICD/OMRSD (Operational Maintenance Requirements and Specifications Document) requirements and typical values.

The replenish phases of loading are initiated after the tanks are filled. Replenishment maintains the tanks at the required propellant level by continuously replacing any propellant that is drained or lost by boiling and venting. The propellant levels are maintained by the LPS, which monitors the discrete wet/dry signals from the liquid level sensors in the ET. (The 100 percent sensors are used unless a failure occurs.) These discrete wet/dry signals are converted to percent wet values by a software program. The replenish valves in each fill system are then modulated to maintain the percent wet within the OMRSD limits required to control the flight propellant mass.

2.2.4 LO_2 Loading System

2.2.4.1 General Discussion of System/Requirements

The LO_2 loading sequence at KSC requires 4 hours from start of facility chilldown through completion of replenish and ready for launch. See figure 2.2-1 for the normal loading time line. Facility chill, orbiter chill, and fill to the 2 percent level are accomplished at fill flow rates of up to 320 gal/min. After the 2 percent level is reached, the LPS initiates fast fill at approximately 1400 gal/min and at the same time initiates an ogive stabilization program. This program controls LO_2 tank ullage pressure between 2.0 and 8.0 psig by operating the vent valve open and closed at appropriate ullage pressure set points. The LO_2 tank ogive requires an internal pressure during fill to pressure-stabilize the tank skin to preclude buckling because of compression loads. After the 98 percent sensors indicate covered, fill flow rate is reduced to topping rate of 240 gal/min and then, when the sensors at 100 percent indicate covered, replenish control is established to maintain the liquid level at 100 percent. The limits and typical values for flow rate, temperature, and pressure at the orbiter/ground servicing equipment (GSE) interface are found in table 2.2-II for each of these phases. During replenish of the early ET's, 2.0 psig ullage pressure was maintained with LPS program to close the vent valve at 2.2 psig and then open the valve after 3 minutes. Now, portions of the ogive have been strengthened to allow for replenish with no pressure control to achieve optimum LO_2 density during replenish. Figure 2.2-8 represents LO_2 tank ullage pressure during the entire loading sequence comparing observed pressures to those predicted by a math model, which is a flow-thermal model of the entire loading system. Correlation is within 8 percent. Because of confidence in the model, various changes in system variables, such as fill rates, heat leak, weather, and design changes may be assessed to determine affects on loading time line as well as on final ullage pressure and liquid density.

Between T-9 and T-4 minutes (depending upon mission requirements) replenishing is terminated by closing the orbiter inboard fill and drain valve and LO_2 drains back through the SSME bleed system. Cold LO_2 from the ET provides LO_2 at engine start temperature requirements. For ET's with the anti-geyser line, the circulation loop results in cold LO_2 (168° R) at the ET-orbiter interface during drainback. The drainback flow rate is about 16.5 lb/sec of LO_2 during the unpressurized period and 18.4 lb/sec during the pressurized period. For LWT's, the drainback flow temperature increases just prior to engine start as the warm propellant in the ET feedline arrives at the orbiter-engine interface. This increase has been determined to be acceptable to the engine.

2.2.4.2 Description of OMI S1003 - MPS LO_2 System Automatic Load and Drain Operations

A. Objective

To fill and drain the LO_2 portion of the orbiter/ET main propulsion system for space shuttle vehicle (SSV) testing and flights.

B. Description

This test includes hazardous operations resulting from the transfer of LO_2 from the pad storage tank to the launch vehicle, plus pressurization of the storage tank and ET. Additionally, inert atmospheres are created in TSM, payload bay (PLB), aft fuselage, and forward reaction control system (FRCS) in conjunction with GN_2 purges required during the loading and drain operations.

C. LO_2 Loading, Replenish, and Drain Sequences

1. Transfer Line and Vehicle Childown

   Included in this sequence are the initial childown of the cross-country piping with childown gases venting from the ET vent valve and facility vent system; maintenance of storage tank pressure at 10 psig; and transfer pump motor start sequence. ET helium bubbling, orbiter/ET anti-icing purges, ET anti-icing purges and T-0 umbilical purge are initiated. Engine purge sequences are started concurrently with this sequence.

2. Orbiter MPS Childown

   Thirteen minutes after starting transfer line childown, the initial delivery of LO_2 to the orbiter MPS and SSME systems is begun. The LO_2 prevalves are verified open (verifying engine purge sequence 3), engine bleed valves are opened, and the transfer pump ramped to 1000 rpm for 2 minutes and then to 2850 rpm. Main fill and TSM drain valves are closed and replenish valve is opened to obtain an initial orbiter flow rate of 250 gal/min. When orbiter/inlet pressure reaches 34 psig, the replenish valve is closed to allow the feedline to drain through the TSM vent. The replenish valve is re-opened and the TSM vent is closed when the orbiter inlet pressure is 10 psig.

3. Slow Fill of ET to 2 Percent

   When three of four LO_2 engine cutoff sensors indicate wet, the TSM vent valve is closed and the replenish valve is opened to permit 270 gal/min to the ET 2-percent level. An 11-minute timer is started.

4. Maintaining LO_2 ET Tank Pressure During Fill Above 2 Percent

   During fill, the pressure will be maintained between 2.2 and 8.0 psig when the LO_2 level is greater than 2 percent.

5. Fast Fill to 98 Percent

   Completion of the 11-minute timer results in the initiation of the fast fill sequence which increases the flow rate to approximately 1350 gal/min. See figure 2.2-9 on LO_2 loading flow during fast fill and replenish.

6. Topping Flow to 100 Percent

   Wetting of the 98 percent sensors terminates the fast fill sequence and reduces pump to 3100 rpm resulting in a flow rate of approximately 260 gal/min.

7. Replenish

As the ET liquid level approaches 100 percent, the MLP replenish valve is throttled to maintain LO_2 at flight mass until the terminal sequence is initiated. Engine conditioning through a LO_2 overboard bleed continues. See figure 2.2-9.

8. Terminate Replenish

At the start of the drainback period, the replenish valve is commanded closed, the inboard fill and drain valve is closed and feedline drain is initiated. LO_2 engine bleed continues to ensure acceptable SSME inlet temperatures at main engine ignition. At T-2 minutes 55 seconds, the LO_2 tank is pressurized to flight pressure. At T-48 seconds, the outboard fill and drain valve closes and numerous GSE purges are secured followed by the opening of both pogo recirculation valves at T-12.5 seconds and by the closure of the over-board bleed valve at T-9.4 seconds. The LO_2 system is in configuration for engine ignition. See figure 2.2-10.

9. Drain

The ET LO_2 tank is pressurized, the storage tank vented, and GSE valving configured for drain. The drain sequence is terminated when the orbiter inlet temperatures indicate greater than -280° F 1 minute after ECO sensors indicate dry. Purging of the ET and orbiter will continue for system warmup and inerting.

The above described sequences will be conducted using goal programs for both the ground and airborne functions, in addition to programs to implement revert/stop flow, and recycle conditions. Manual (keyboard/payload function key (PFK)) and/or hardwire control will be exercised in the event contingency conditions dictate.

2.2.4.3 Performance Perspective on Drainback

The liquid level at the initiation of drainback is the same for all flights. The zero σ level is that fluid level required for a nominal mission. The 3 σ level is that amount of fluid on "reserve" for a 3 σ bad day with respect to performance. The zero σ and 3 σ levels are mission dependent. See figure 2.2-11.

For extended holds after drainback has been initiated, the margin level may go to zero and some of the FPR may not be available. Once the FPR is affected the decision to fly is left up to management and LSEAT.

Any loss in performance that occurs prelaunch is input into the ARD. This loss in performance will cause the abort boundaries to occur later than the nominal times.

The performance associated with drainback can be calculated using the following equation:

$$\text{Performance Penalty} = \underbrace{(0.23221 \times \Delta LO_2)}_{LO_2 \text{ Term}} - \underbrace{(.9337 \times \Delta LH_2)}_{LH_2 \text{ Term}} - \Delta FPR \qquad (1)$$

The coefficients in the LO_2 and LH_2 terms are calculated values that take into account the propellant density and overboard mixture ratio. However, during drainback, there should be no loss of LH_2, so the ΔLH_2 term will be zero. The loss of LO_2 (ΔLO_2) during drainback is calculated by multiplying the drainback flowrate (18.17 lb/sec) by the time (in seconds):

$$\Delta LO_2 = 18.17 \text{ lb/sec} \times \text{time (seconds)} \qquad (2)$$

The next step is to adjust the fuel bias. The typical nominal fuel bias for a 4:55 drainback period is 968 lb of fuel. The FPR value is obtained from an FPR table. The typical nominal FPR value is 4698 lb. Since some LO_2 mass has been lost, there is a corresponding increase in fuel bias. This is determined by dividing the ΔLO_2 by the overboard mixture ratio:

$$\Delta \text{ fuel bias} = \Delta LO_2/MR = \Delta LO_2/6.034 \qquad (3)$$

This change in the fuel bias must be added to the nominal fuel bias to obtain the adjusted fuel bias

$$\text{Adjusted fuel bias} = 968 \text{ lb } LH_2 + \Delta \text{fuel bias} \qquad (4)$$

This adjusted value is used to obtain the new FPR value (FPR') from the FPR table. The change in FPR (ΔFPR) is calculated by subtracting the FPR term from the nominal FPR term:

$$\Delta FPR = 4698 - FPR' \qquad (5)$$

The total performance penalty is then determined by substitution into equation 1.

18.17 lb/sec = drainback rate LO_2 TERM - LH_2 TERM - ΔFPR

I. Usable propellant (perf. penalty) = $(.23221 \times \Delta LO2) - (.9337 \times \Delta LH_2) - \Delta FPR$*

* From FPR table (968 Fuel Bias). $\Delta FPR = (4698 - \alpha)$

II. ΔLO_2 = 18.17 #/sec x time in drainback hold (seconds)

III. For a decrease in LO_2, the corresponding LH_2 bias will increase

$\frac{\Delta LO_2}{OV.MR} = \frac{\Delta LO_2}{6.034}$ = increase in fuel bias (ΔFPR)

IV. Adjusted fuel bias = 968 + ΔFPR = use new fuel bias to get new value from FPR table = α

V. $4698-\alpha$ = FPR Hit

VI. Total performance penalty = $(.23221 \times \Delta LO_2)$ - FPR hit

Example: 2 min drainback/hold; assume $\Delta LH_2 = 0$

I. Perf penalty = [.23221 X 18.17lb/sec (120 sec)] - [4698 $-\alpha$]

II. $\Delta LO_2 = (18.17 \frac{lb}{sec})(120\ sec)$

$= 2180$ lb

III. $\frac{\Delta LO_2}{OV.MR} = \frac{2180\ lb}{6.034}$ = 361 lb LH_2

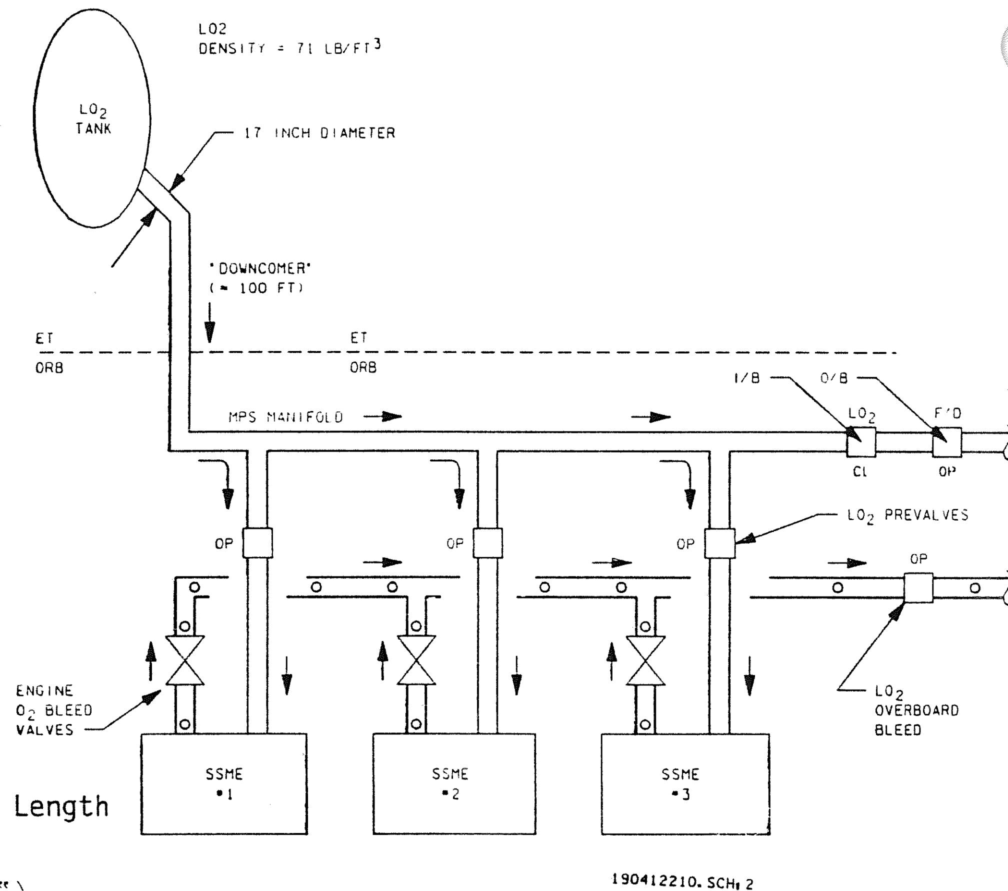
IV. ΔFPR = 968 + 361 = 1329 lb LH_2 new fuel bias

new FPR value from FPR table for 1329 lb fuel bias = 4360 = α

V. FPR hit = 4698 - 4360 = 338 lb

VI. Perf. penalty = .23221 x 2180 lb - 338 lb

= 168 lb



Volume = Diameter x Length

$$V = \pi\left(\frac{17''}{2}\right) x\, 100'\left(\frac{12''}{1}\right)$$

V = 272238 in^3 = 157.5 ft^3

LO_2 Mass = 157.5 ft^3 (71 lb/ft^3)

= 11,186 lb

Total time LO_2 drainback in downcomer = $\frac{LO_2 \text{ mass}}{18.17 \text{ lb/sec}}$

= $\frac{11186 \text{ lb}}{18.17 \text{ lb/sec}}$

= 616 seconds

= 10.3 minutes

Normal drainback hold time = 4:55 = 4.92 minutes

Therefore, a hold of 10.3 min - 4.92 min = 5.38 min

will deplete the warmer fluid in downcomer

2.2.5 LH_2 Loading System

2.2.5.1 General Discussion of System/Requirements

The LH_2 tank is loaded in a 3-hour timeline with special controls to achieve enhancement of the ET super light ablator (SLA) which is part of the TPS. This enhancement is similar to pre-stress of materials such as concrete. The tank is pre-pressurized before LH_2 enters the tank so that the TPS material is "stretched" prior to being chilled to the LH_2 temperatures. The material maintains its memory after chilldown so that, when flight operating pressures are applied, the TPS does not experience high stresses caused by internal tank pressures. The loading sequence consists of venting the tank and initiating facility chill, closing the vent valve, and pressurizing the tank to enhancement pressure slow fill to 2 percent, fast fill at 8300 gal/min from 2 percent to 85 percent, slow flow at 1100 gal/min from 85 percent to 98 percent, topping at 560 gal/min from 98 percent to 100 percent, and replenish at 100 percent. At 98 percent the vent valve is opened (at this time TPS enhancement is complete) and chilldown to steady state ullage pressure during replenish is achieved. Enhancement pressure is maintained by opening the vent valve at 32 psig and closing at 29 psig during the fill sequence. The entire logic to initiate fill, cycle the vent valve, transition fill rates, and control replenish is accomplished automatically by the LPS. For future ET's, the TPS design is optimized so that fast fill can be accomplished from 2 percent to 98 percent and other logic changes will be made so that the fill timeline will be 2 hours. (See Future Improvements.) Figure 2.2-12 represents LH_2 tank ullage pressure during the entire loading sequence. Various changes in system variables, such as the time line reduction discussed in Future Improvements, can be assessed using data from the first flights and a math model similar to the LO_2 model.

Before the start of LH_2 propellant loading, the tank is vented to ambient pressure to verify accuracy of the ullage pressure transducers. The intertank GN_2 purge is initiated to remove air and achieve an inert intertank environment before propellant loading. Oxygen (O_2) concentration decay rate is a function of the GN_2 purge flow rate, intertank volume and intertank vent area.

Terminal count is initiated at T-9 minutes. There is a 10 minute hold at this point. At this time, there is a final verification that propellant load is within "flight mass OK" limits, which are determined by the 100 percent sensors indicating 10 percent to 70 per-cent wet for LH_2.

At T-1 minute 57 seconds, the LH_2 tank vent valve is closed; and, at T-1 minute 35 seconds, helium prepressurization is initiated and monitored for "signature" trace. A signature trace of ullage pressure is a function of ullage volume, prepressurization flow rate, and heat transfer effects on ullage gas. Analytical prediction of the signature compares well with actual ullage pressure data.

The launch commit criteria (LCC) for the LH_2 tank ullage transducer is based on precluding in-flight venting and assuring ullage pressure is maintained above 32 psia in-flight to provide required NPSP for the SSME's. An LCC analysis of the prepressurization signature indicates normally six cycles will occur before T-31 seconds. A valve leak rate of 0.2 lb/sec would result in nine cycles. In order to preclude leakage above 0.2 lb/sec, a launch scrub/delay would occur automatically by the LPS if nine or more pre-pressurization cycles occurred. On each of the first four flights, six cycles have been recorded.

2.2.5.2 Description Of OMI NO. - S1004 - MPS LH_2 System Automatic Load and Drain Operations

A. Objective

1. To fill the LH_2 portion of the orbiter/ET main propulsion system to the required flight mass for SSV testing and flights.

2. To drain the SSV in support of loading tests, CDDT, launch abort or scrub.

B. Description

This test includes hazardous operations resulting from the transfer of LH_2 from the pad storage tank to the launch vehicle and pressurization of the ET during terminal sequence and drain operations. Additionally, inert atmospheres are created in TSM, PLB, aft fuselage, FRCS and ET intertank in conjunction with GN_2 purges required during loading and drain operations.

C. Liquid Hydrogen Loading, Replenish, and Drain Sequences

1. Facility/Orbiter Chilldown

This sequence starts a liquid chilldown of the storage facility, cross-country piping, orbiter, and ET with chilldown gases delivered to the burn pond via ET vent system. Prior to initiating chilldown, SSME purge sequence 3 is verified, ensuring the prevalves and engine bleed valves are open.

2. Storage Tank Pressurization and Slow Fill

Four minutes after initiating chilldown, the LH_2 storage tank is pressurized to 66 psig and LH_2 slow fill commences.

3. Fast Fill to 98 Percent

Three of four ECO sensors wet causes the start of the fast fill sequence at a flow rate of approximately 8,000 gal/min. The MPS topping valve and high point bleed valves are opened at start of fast fill. LH_2 recirculation pumps are started 25 minutes after start of fast fill and RECIRC valves are opened and prevalves are closed. At approximately 85 percent, flow rate is reduced to approximately 1400 gal/min.

4. Topping Flow to 100 Percent

   Wetting of the 98 percent sensors terminates the fast fill sequence resulting in a reduction in flow rate to approximately 1000 gal/min. Flow continues from the facility through the topping valve to the 100 percent level.

5. Replenish at 100 Percent

   The MLP replenish valve throttles to maintain the LH_2 tank at flight mass until terminal sequence is initiated. High point bleed continues through the overboard vent.

6. Terminate Replenish

   At T-1 minute 57 seconds, the LH_2 replenish valve is commanded closed, followed by closure of the MPS topping valve and drain assist purge is initiated. Pressurization of the ET LH_2 tank is initiated when the topping valve closes and ET vent line purges are initiated. The outboard fill and drain valve closes at T-46 seconds; and, at T-9 seconds, recirculation is terminated by opening the pre-valves, turning off the recirculation pumps, and closing the recirculation valves. The high point bleed valve is closed at T-9 sec. This places the LH_2 system in configuration for engine ignition.

7. Drain

   The ET LH_2 tank is pressurized, the storage tank vented, and GSE valving configured for drain. The drain sequence is terminated after the ET low level sensors indicate a dry condition followed by purging of the ET and orbiter for system warmup and inerting. Approximately 300 gallons of LH_2 that must be depleted by boil-off will be left in the ET following drain.

The above described sequences will be conducted using goal programs for both the ground and airborne functions, in addition to programs to implement revert and revert-avoidance conditions. Manual (keyboard/PFK) and/or hardwire control will be exercised in the event contingency conditions dictate.

TABLE 2.2-I.- CRYOGENIC LIQUID PROPERTIES

PROPERTIES OF LIQUID OXYGEN

Item	Procurement limits	Delivered to interface *
Purity	99.6 percent by volume (maximum)	99.2 percent by volume (minimum)
Alkyne as acetylene	0.25 ppm by weight (maximum)	1.55 ppm by weight (maximum)
Total hydrocarbons	50.0 ppm by volume as methane (maximum)	75.0 ppm by volume as methane (maximum)
Moisture	3.0 ppm by volume (maximum)	26.3 ppm by volume (maximum)
Particulate	1.0 mg/1 (maximum) liquid only	
Density **	70.8 Lb/ft3	

PROPERTIES OF LIQUID HYDROGEN

Item	Procurement limits	Delivered to interface *
Total purity	99.995 percent by volume (minimum)	99.994 percent by volume (minimum)
Total gaseous impurities	50 ppm by volume (maximum)	60 ppm by volume (maximum)
Selected impurities (nitrogen, water and volatile hydrocarbons)	9.0 ppm by volume (maximum)	9.0 ppm by volume (maximum)
Specific impurities		
Oxygen plus argon	1.0 ppm by volume (maximum)	5.0 ppm by volume (maximum)
Helium	39.0 ppm by volume (maximum)	45.0 ppm by volume (maximum)
Carbon monoxide plus carbon dioxide	1.0 ppm by volume (maximum)	1.0 ppm by volume (maximum)
Density **	4.43 lb/ft3	

*Orbiter/ET interface
**Saturated liquid at one atmosphere pressure

TABLE 2.2-II.- LO_2 AND LH_2 LOADING PARAMETERS

Line and number	Function	Flow direction	Media	Operating flow rate, lbs/sec			Temperature, °R			Operating pressure, psia		
				Minimum	Nominal	Maximum	Minimum	Nominal	Maximum	Minimum	Nominal	Maximum
LO_2 feedline-line	LO_2 Prechill	ORB to ET	GO_2	8	16	32	162	N/A	580	15	25	40
	LO_2 Orbiter chill fill	ORB to ET	GO_2/LO_2	0	32	51	162	168	190	15	30	46
	LO_2 Initial fill to 2 pct.	ORB to ET	LO_2	21.0	32.0	50.9	162	N/A	173	20	N/A	60
	LO_2 Fast fill	ORB to ET	LO_2	174	N/A	790	162	166	170	60	85	105
	LO_2 Replenish	ORB to ET	LO_2	0	5.5	11	N/A	N/A	181	79	82	84
	LO_2 Stabilize	ET to ORB	LO_2	0	6	12	N/A	N/A	181	79	82	84
	LO_2 Steady state replenish	ORB to ET	LO_2	1.20	1.53	1.96	N/A	N/A	180	79	82	84
	LO_2 Drain	ET to ORB	LO_2	300	530	790	162	164	170	20	60	95
	LO_2 Drainback (5 minutes)	ET to ORB	LO_2	16.6	19.6	22.6	163	N/A	178	79	92.5	106
	LO_2 Engine feed	ET to ORB	LO_2	1680	2658	2945	162	N/A	170	20	100	201
	LO_2 Topping	ORB to ET	LO_2	90	N/A	158	N/A	N/A	174	79	82	92
	Tank purge	ET to ORB	GN_2	2.75	N/A	3.10	480	530	580	N/A	N/A	25
	Tank purge	ET to ORB	GH_e	0.25	0.30	0.35	480	530	580	N/A	N/A	25
O_2 Press, Line - line	Ice/frost minimization	ORB to ET	GH_e	0.006	0.009	0.039	498	579	660	N/A	N/A	50
	O_2 Pre-pressurization	ORB to ET	GH_e	0.25	0.30	0.35	480	530	580	60	80	106
	O_2 Pressurization	ORB to ET	GO_2	2.5	4.65	8.1	300	840	960	100	300	600
	Tank purge	ORB to ET	GN_2	2.75	N/A	3.10	480	530	580	N/A	N/A	600
	Tank purge	ORB to ET	GH_e	0.25	0.30	0.35	480	530	580	N/A	N/A	106
	LO_2 Emergency drain pressurization	ORB to ET	GN_2	2.75	N/A	3.10	480	530	580	N/A	N/A	600
LH_2 feedline-line	LH_2 Facility/ORB/ET/chill	ORB to ET	GH_2/LH_2	0	N/A	10	38	N/A	560	15	N/A	45
	LH_2 Fill transition and initial chill	ORB to ET	LH_2	1	N/A	15	37	N/A	N/A	15	N/A	52
	LH_2 Fast fill	ORB to ET	LH_2	75	110	118	36.5	37	38	15	20	52
	LH_2 Drain	ET to ORB	LH_2	50	80	118	36.5	37	40	32	34	36.5
	LH_2 Recirculation	ET to ORB	LH_2	4	4.5	5	36.3	37	37.3	16.9	N/A	48
	LH_2 Engine feed	ET to ORB	LH_2	281	444	492	36.5	37	40	30	35	50
	Tank purge	ORB to ET	GH_e	0.75	1.25	1.5	480	530	580	N/A	N/A	25
	Tank purge	ET to ORB	GH_e	1.2	1.35	1.5	480	530	580	N/A	N/A	25
H_2 Press, Line - line	LH_2 Pre-pressurization	ORB to ET	GH_e	1.2	1.35	1.5	480	530	580	120	150	220
	LH_2 Pressurization	ORB to ET	GH_2	0.45	2.10	4.07	300	525	565	100	300	745
	Tank purge	ORB to ET	GH_e	1.2	1.35	1.50	480	530	580	120	150	220
	Ice/frost minimization	ORB to ET	GH_e	0.004	0.006	0.039	494	577	660	N/A	N/A	50
LH_2 Recircu-lation line - line	LH_2 Replenish	ORB to ET	LH_2	0	2	5	N/A	N/A	38.1	17	18	19
	LH_2 Replenish and recirculation	ORB to ET	LH_2	4	6.5	10	N/A	N/A	38.1	17	18	19
	LH_2 Recirculation	ORB to ET	LH_2	4	4.5	5	N/A	N/A	44.7	17	N/A	48
	LH_2 Topping	ORB to ET	LH_2	5	9	12	N/A	N/A	41	17	18	55
	LH_2 Topping and recirculation	ORB to ET	LH_2	9	13.5	16	N/A	N/A	41	17	18	55

TABLE 2.2-III.- MPS LO_2 SYSTEM AUTOMATIC LOAD OPERATIONS

[Ref. drawing 79 K05735C - LO_2 Operating Criteria]
OMI-51003, Rev. J

LO_2 System Auto Load Operation Only

3-000 LO_2 Loading Preliminaries

3-002 Complete preparation of system per OMI G3151

3-011 Verify A91 storage tank ullage press. 9-11 psig
(tank ullage press. initially at 1.5 psig)

A9	Closed	Storage Tank Vent Valve
A265	Open	Vaporizer Control Valve Enable Solenoid
A12	Cycling	Vaporizer Control Valve
A77	Open	Pressurization Vaporizer Shutoff Valve
A86544	Open	IM Pump Suction Line Chilldown Valve
A86461	Closed	Transfer Line Fill Valve
A86462	Closed	TSM Drain Valve
A196	Open	Bypass Shutoff Valve
A128	Open	Pump Chilldown Valve

ET Ullage Pressure 1.6 to 6.5 psig
ET liquid level <1%

3-015 LO_2 loading preliminaries complete.

4-000 MPS Loading Preliminaries

4-002 Verify OMI V9018 (SSME/MPS & securing) complete

4-003 Verify He pressure is 1800 psia minimum

4-004 Verify SSME's are ready for loading

4-007 Verify:

Open	PV10	LO_2 Inboard Fill & Drain Valve
Close	PV9	LO_2 Outboard Fill & Drain Valve
Close	PV7	LO_2 Feedline Relief Shutoff Valve
Open	PV19	LO_2 Overboard Bleed Valve
Close	PV20	LO_2 Pogo Accum. Recirc. Valve
Close	PV21	LO_2 Pogo Accum. Recirc. Valve

4-011 Initiate SSME purging per seq. 15

4-013 MPS preliminaries complete, ready for LO_2 loading

5-000 Transfer Line Chilldown

TABLE 2.2-III.- Continued

5-002	Verify ET/IT heated purge ON minimum of 30 minutes		
5-005	Initiate auto fill		
5-007	PV10	Open	LO_2 Inboard Fill & Drain Valve
	PV9	Open	LO_2 Outboard Fill & Drain Valve
	PD1	Open	LO_2 Orb/ET Feed Disconnect Valve
	PV7	Closed	LO_2 Feedline Relief Shutoff Valve
	PV20	Closed	LO_2 Pogo Accum. Recirc. Valve
	PV21	Closed	LO_2 Pogo Accum. Recirc. Valve
5-008	A105967	ON	Anti-icing Purge (Pressurization Line)
	A75074 >15 psig and <100 psig - ET pre-press. output		
5-009	A105958	ON	Press. Anti-icing Purge Heater
5-011	A75090	ON	Umbilical Carrier Purge
5-012	A105987	ON	ORB/ET Disc. Purge
5-015	A86461	Open	Transfer Line Fill Valve
	A86460	Open	Replenish Valve
	A86483	Open	TSM Vent Valve
	A86462	Open	TSM Drain Valve
		Open	ET LO_2 Vent Valve
	A86927	He Bubbling Differential Pressure >.65 and <1.9 psid	
5-017	A86461 will be closed if ET ullage exceeds 12 psig and A86460 will be closed if ET ullage exceeds 14 psig. Both valves are reopened when pressure drops below 5 psig.		
5-018	A141	Open	Pump Suction Valve
5-019	Beginning 5-Minute Timer		
5-020	A86544	Closed	Pump Suction Line Chilldown
	A129	Open	Pump Chilldown Valve
	A134	Close	Transfer Line Drain Valve
5-021	At 5-minute timer complete pump motor will be started		
5-022	Verify A126 Motor ON		
5-023	Pump Motor ON beginning 5-minute timer		
5-024	A105958 >200° F		Anti-icing Purge Temp
	A78160 or A78161		3000+300 psig Intertank Purge Pressure
	Intertank Temp 33° to 118° F (T41T1810H)		

TABLE 2.2-III.- Continued

5-025 With one minute of above 5-minute timer remaining, verify

A86461	Open	Transfer Line Fill Valve
A86460	Open	Replenish Valve

5-026 SSME bleed valves greater than 80%

PV1	Open	Eng #1 LO_2 Prevalve
PV2	Open	Eng #2 LO_2 Prevalve
PV3	Open	Eng #3 LO_2 Prevalve
PV19	Open	Orb. Overboard Bleed Valve

5-028 Transfer line chilldown complete
Starting orbiter MPS chilldown

6-000 <u>Orbiter MPS Chilldown</u>

6-002 Monitor for

A86483	Open	TSM Vent Valve
A86462	Open	TSM Drain Valve
A86461	Open	Transfer Line Fill Valve
A86460	Open	Replenish Valve

6-003 Replenish valve open, increasing pump speed to 1000 rpm

6-004 A126/A127 pump speed ramps to 1000 ±100 rpm

6-005 Operate pump at 1000 rpm for 2 minutes

6-006 After 2 minutes increase pump speed to 2850 rpm

6-007 A128 Closed Pump Chilldown Valve
A126/A127 Pump speed ramps to 2850+/-100 rpm

6-008 Operate pump at 2850 rpm for 2 minutes

6-009 Orbiter inlet press. <140 psig

6-010

A86461	Closed	Transfer Line Fill Valve
A86460	70%	Replenish Valve
A86462	Closed	TSM Drain Valve

6-012 Orbiter Inlet Press. >34 psig or
Skid Outlet Press. >56 psig

6-013 A86460 Closed Replenish Valve

6-014 Begin 12-minute timer for pressure check

TABLE 2.2-III.- Continued

6-015	Monitor for Orb. inlet <10 psig Skid outlet press. <32 psig		
6-017	A86460 A86462 A86461	Open Open Open	Replenish Valve TSM drain valve Transfer line fill valve
6-018	Begin 3-minute timer		
6-019	A86461 A86462	Close Close	Transfer line fill valve TSM drain valve
6-020	Monitor for Orb. inlet press. >10 psig or skid outlet press. >32 psig		
6-021	Orbiter childown complete--Begin slow fill, ogive stabilization inhibited		
7-000	LO_2 Slow Fill to 2 Percent		
7-002	Monitor: Orbiter Inlet Temp (A86471) Manifold Disc. A, B Feedline No. 3 Feedline No. 2	 >-291° F >-284° F >-285° F >-289° F	Time (sec) 30 20 20 20
7-003	A86483 A86460 ORB inlet press <140 psig ET vent valve open	Closed Open	TSM Vent Valve Replenish Valve
7-004	Begin 11-minute slow fill timer		
7-005	Monitor replenish flowrate of 270 ± 20 GPM from 3 minutes. after start of slow fill to end.		
7-009	Monitor for redline temp. violations		
7-011	Slow fill timer complete		
7-012	A86460	Open	Replenish valve
7-013	Ogive stabilization active		

TABLE 2.2-III.- Continued

7-014 A78410, 411, 412 On Helium Bubbling
A105967 On Anti-icing Purge
When ET low ullage press. xducers <2.2 psig, close ET LO_2 Vent Valve.
When ET ullage press. >8.0 psig, open ET LO_2 Vent Valve.

8-000 <u>LO_2 Fast Fill to 98 Percent</u>

8-001 Monitor and notify if redline violation

8-002 Slow fill complete - transition to fast fill

8-006 A86461 Open Transfer Line Fill Valve
A196 Close Bypass Shutoff Valve

8-007 Increase pump speed to 3450 rpm (A126)
Nominal flow 1365 gal/min
Max <1400 gal/min

8-009 In stable LO_2 fast fill to 98%

8-010 Verify solid wet ECO's

8-015 Fast fill will terminate and topping will be initiated automatically when either of the 98% sensors are 50% wet or when the 99.85% is 15% wet.

8-016 Fast fill complete; decrease pump speed to 3100 rpm for ET topping.
Ogive stabilization inhibited

8-018 Control nose cone heater by monitoring:

A106532 >290° F Pri Nose Cone Htr Pge Temp.
A106533 >290° F Sec Nose Cone Htr Pge Temp.

9-000 <u>LO_2 Topping to 100 Percent</u>

9-002 Pump rpm decreased to 3100+/-100

A196 Open Bypass Shutoff Valve
A86460 Open Replenish Valve
A86461 Close Transfer Line Fill Valve

9-003 Open LO_2 vent valve

9-004 Topping to 100% in progress

9-005 Monitor temps for redline violation: Orb. inlet, eng. inlets, manifold disc., bot & mid feedline.

9-006 Perform PV19 (overboard bleed valve) cycle test if required.

TABLE 2.2-III.- Concluded

9-015 Topping will be terminated and replenish will be automatically initiated when any one of the two 100% or 100.15% sensors are 10% wet.

9-016 Topping complete; call replenish

10-000 <u>LO_2 Replenish at 100 Percent</u>

10-002 Begin Replenish Sequencer execution - model adjusts flowrate into the ET to maintain 100% level.

Ogive stabilization inhibited.

10-004 Pump rpm reduces to 2850+/-100 rpm

A86460	15% open;	Replenish Valve
	Open	ET LO_2 vent valve

10-005 Eng Bleed Valves >80% Open

10-006 A86460 modulates flow 0-250 gal/min
Monitor Flight Level OK - 100% Sensor = 15 to 75% Wet
Monitor for 100% Sensor failure >99% Wet
Monitor overfill. When 101% sensor >20% wet, stop flow occurs automatically.

10-007 Monitor for temp. redline violations

10-010 Stable replenish - verified with vent valve open, 100.15% sensor <50% wet and 99.85% sensor >50% wet.

10-011 Verify Flight Mass OK
Monitor temperature redlines

11-000 <u>Terminal Count Sequence/Lift-off</u>

11-002 Verify flight level OK

11-004 TO -8m 57s Begin terminal count sequence

11-005

Initiate LO_2 drainback

11-005	A86460	Close	Replenish Valve
11-006	PV10	Close	LO_2 Inboard Fill & Drain

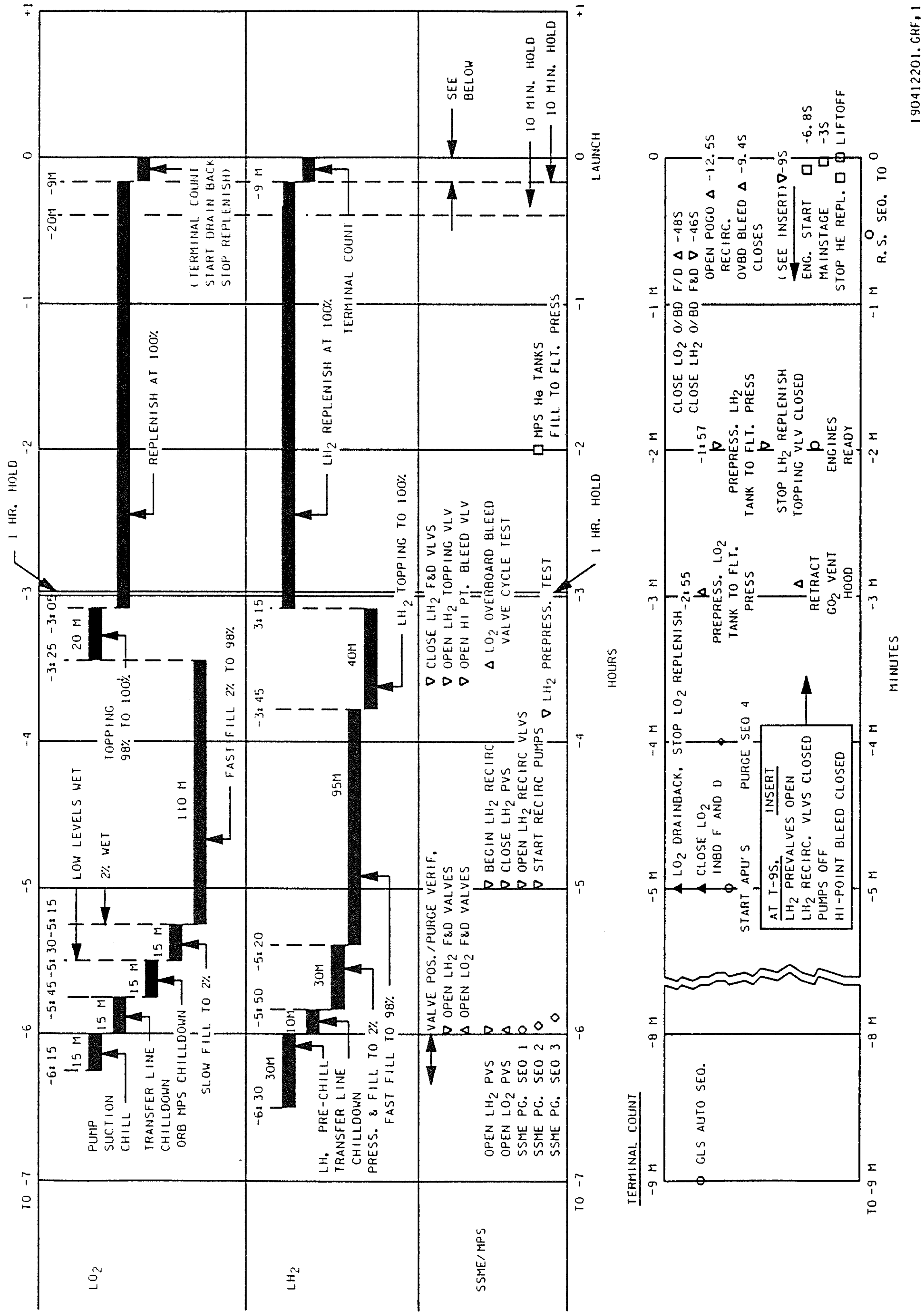


Figure 2.2-1.- MPS propellant loading time line.

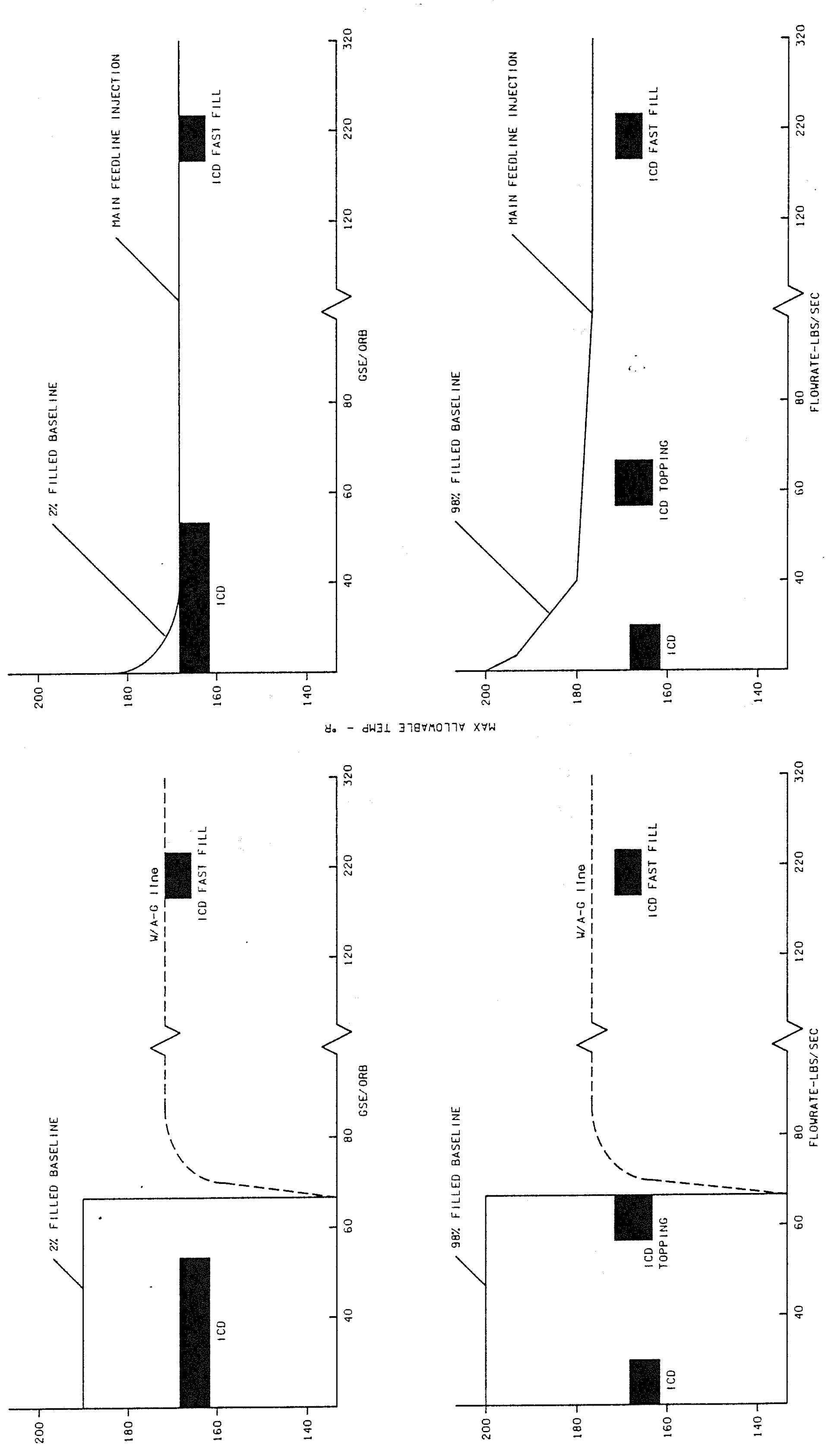


Figure 2.2-2.- Allowable temperature limits to preclude geysering with and without antigeyser line.

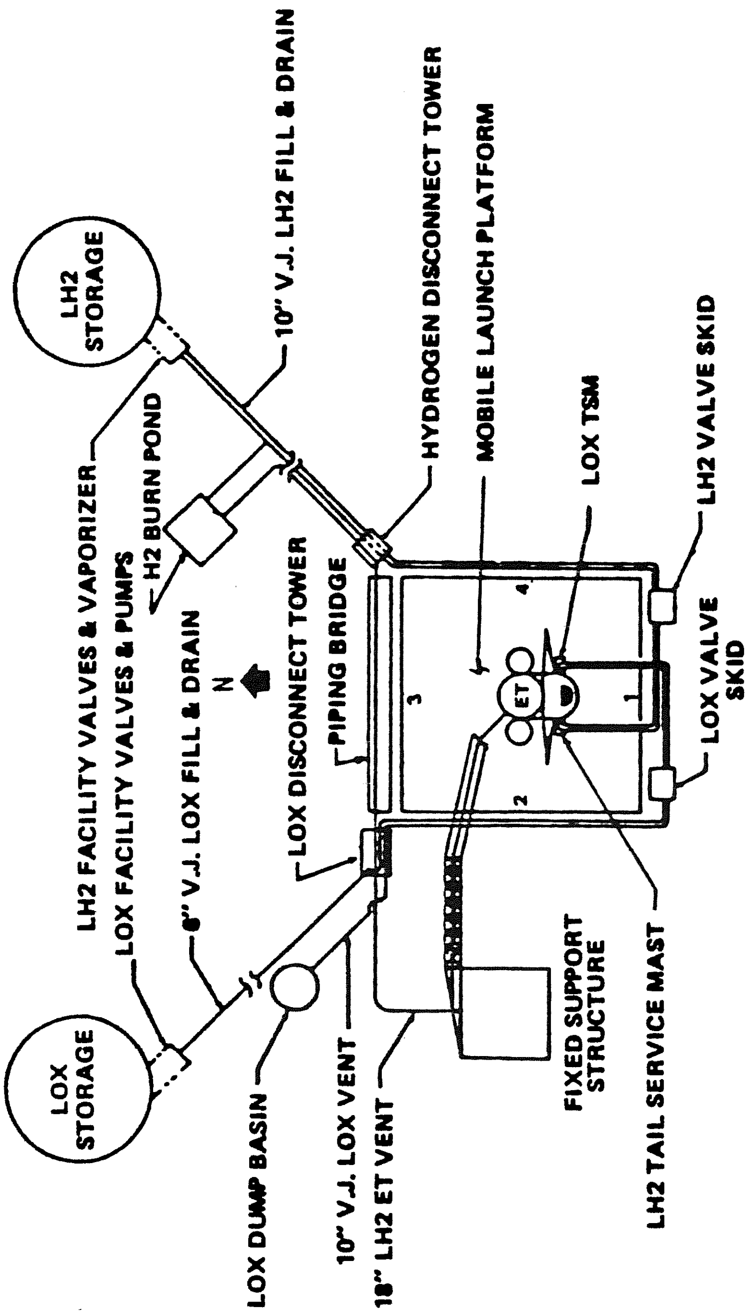


Figure 2.2-3.- Propellant loading facility plan view.

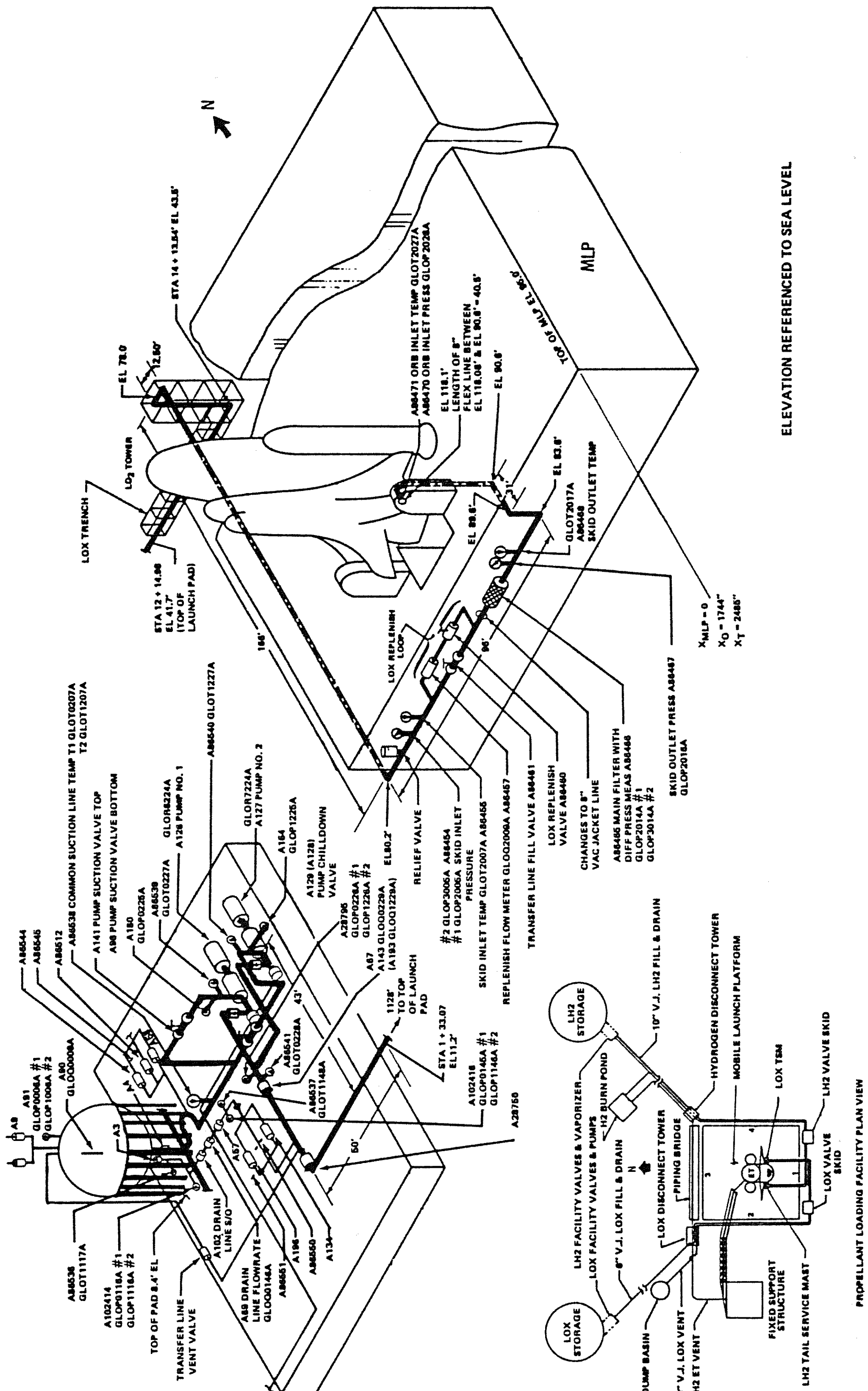


Figure 2.2-4.- LOX loading sketch.

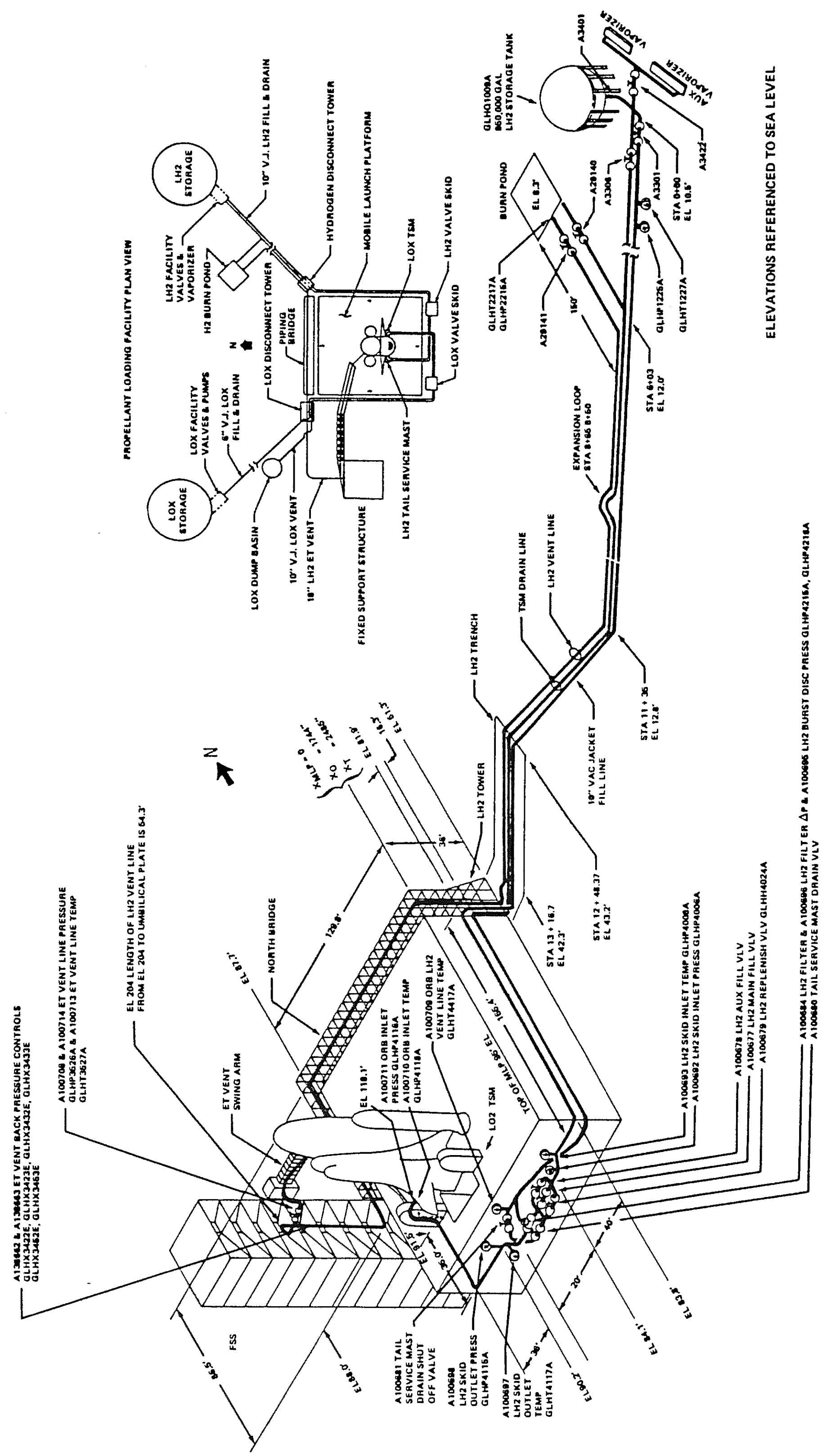


Figure 2.2-5.- LH2 loading sketch.

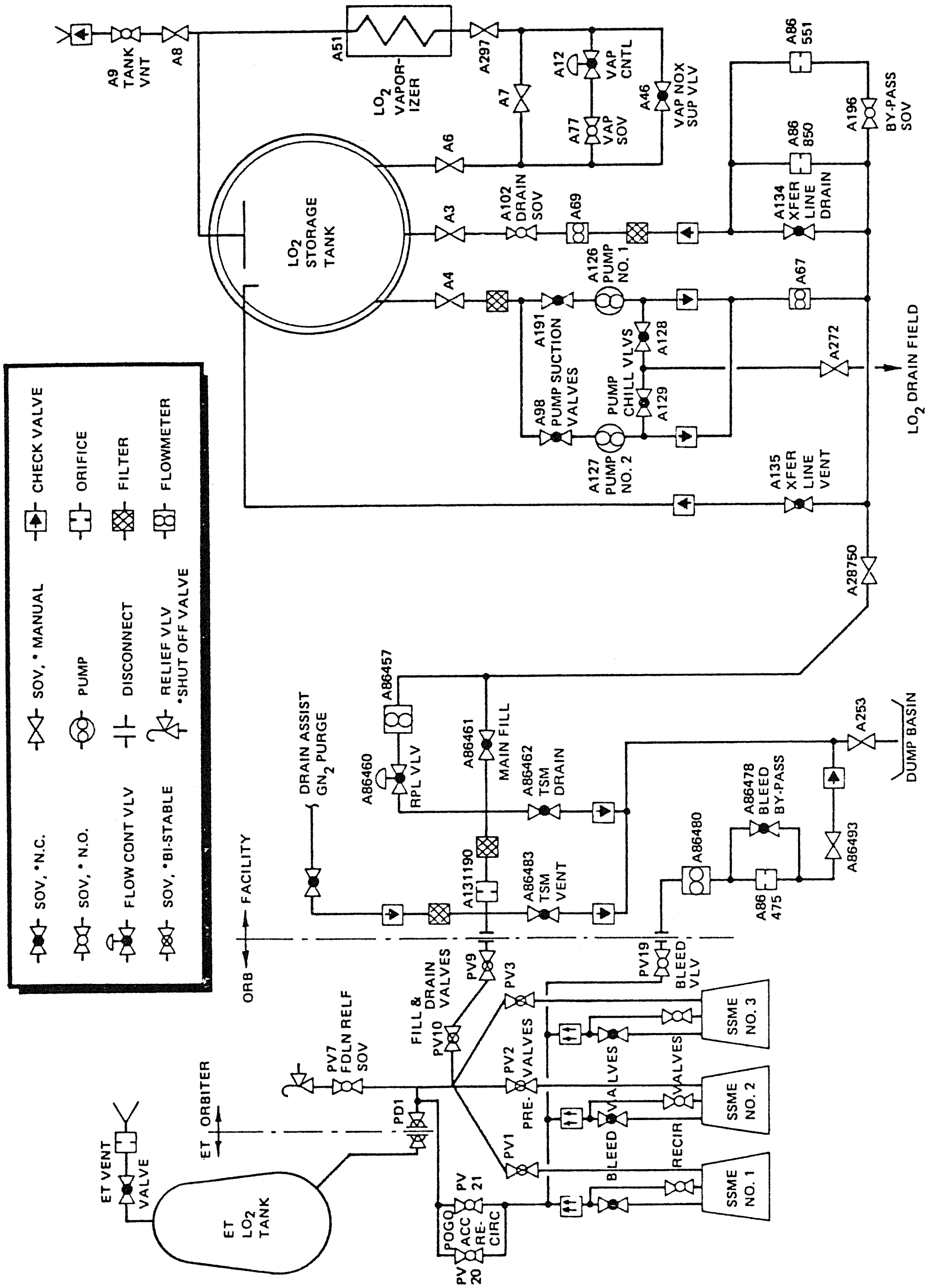


Figure 2.2-6.- KSC LO2 schematic.

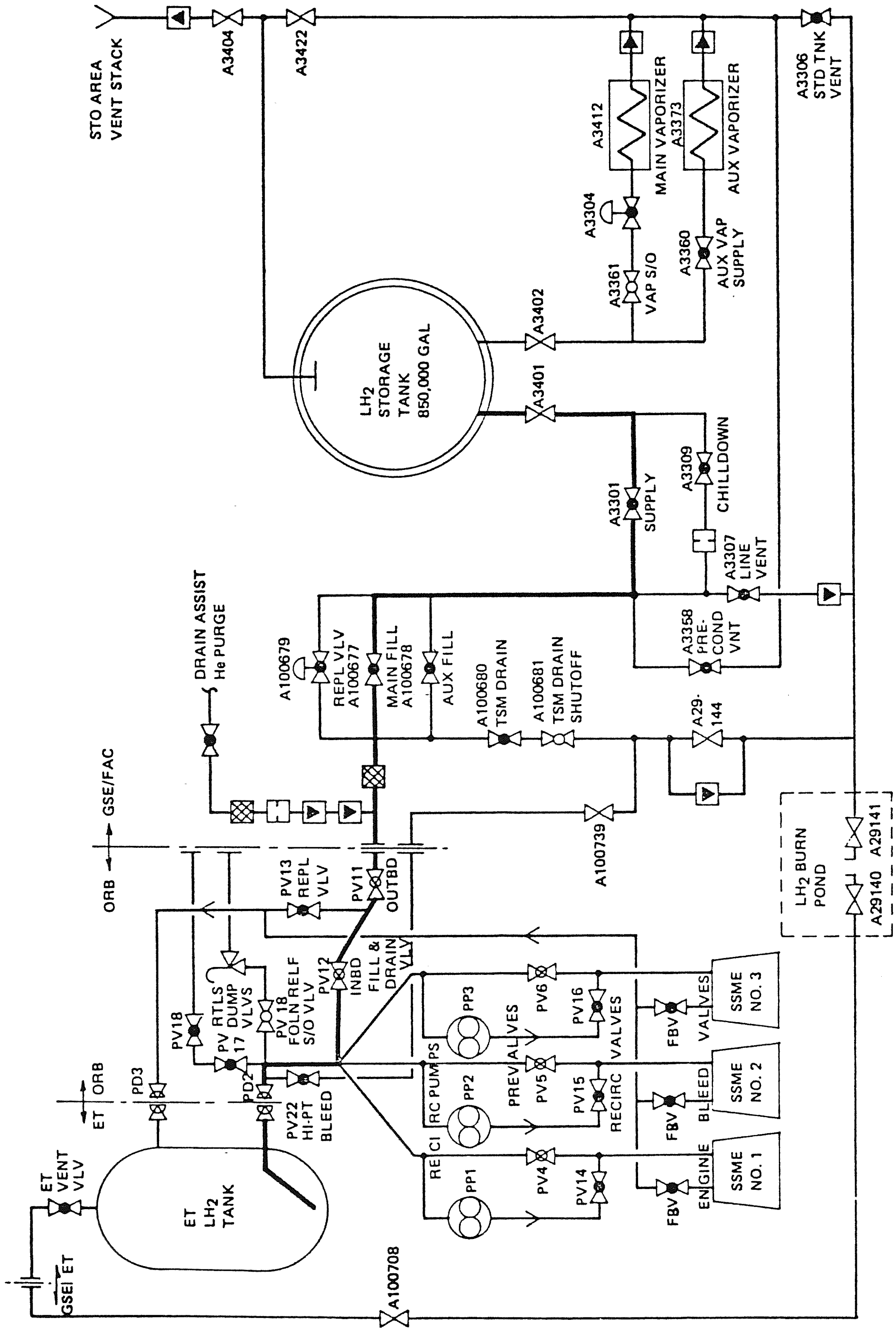


Figure 2.2-7.- KSC LH2 schematic.

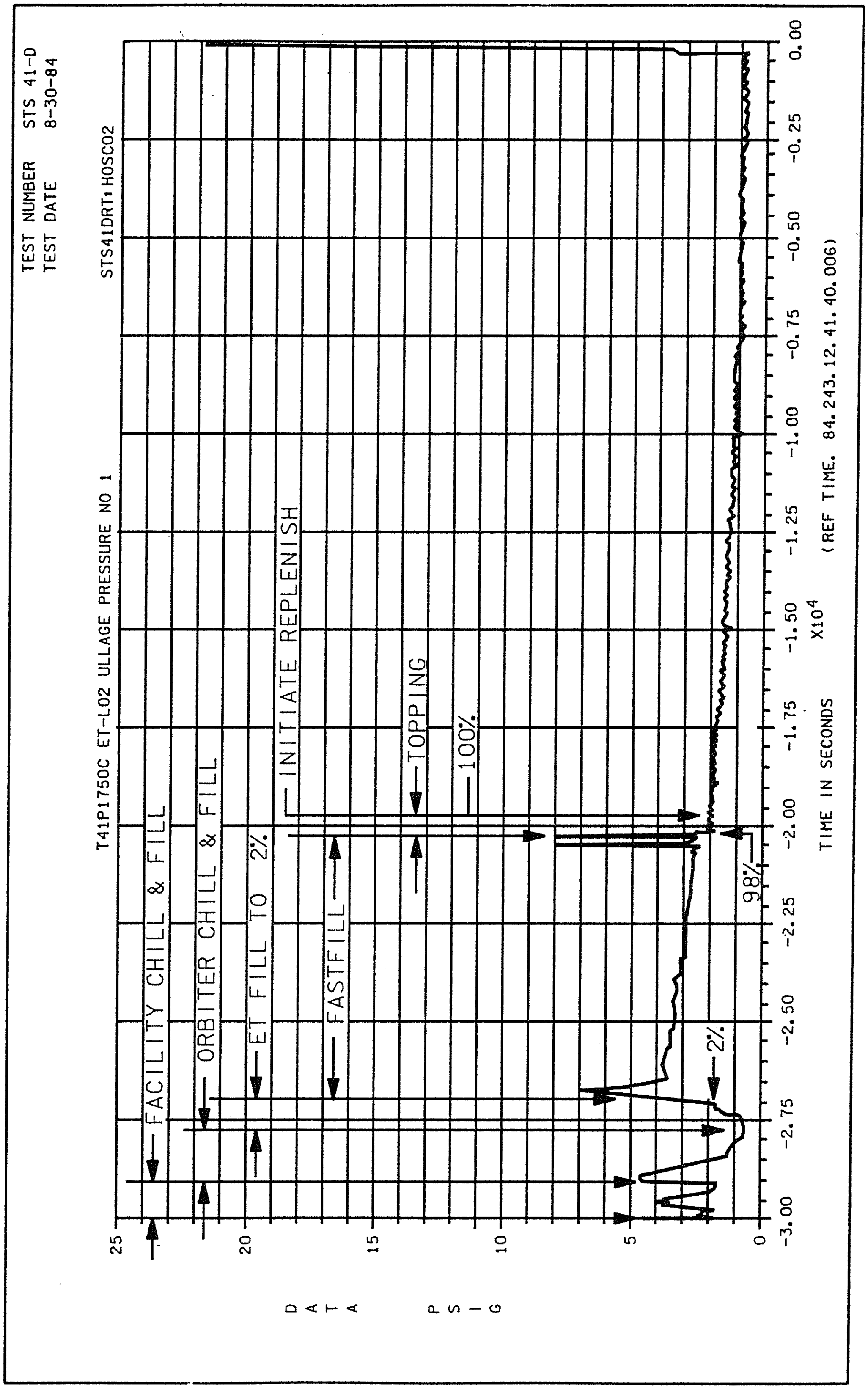
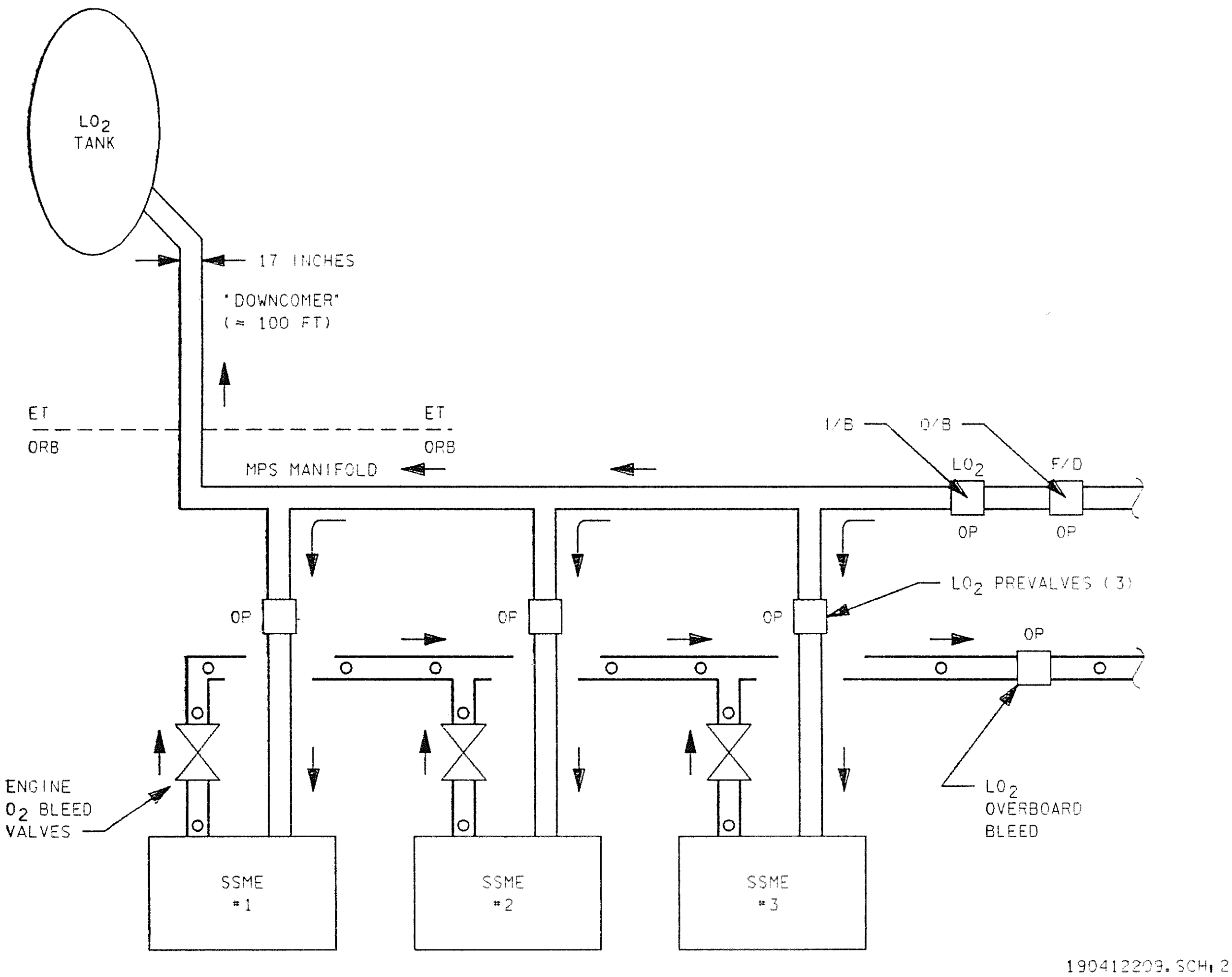


Figure 2.2-8.- LO2 ET ullage pressure.



During the fast fill and replenish, the inlet temperatures are equal to the LO_2 temperatures (-300° F).

LCC is - 287.7° F maximum; continuous verify for 30 minutes prior to terminating replenish.

Figure 2.2-9.- LO_2 loading flow during fast fill and replenish.

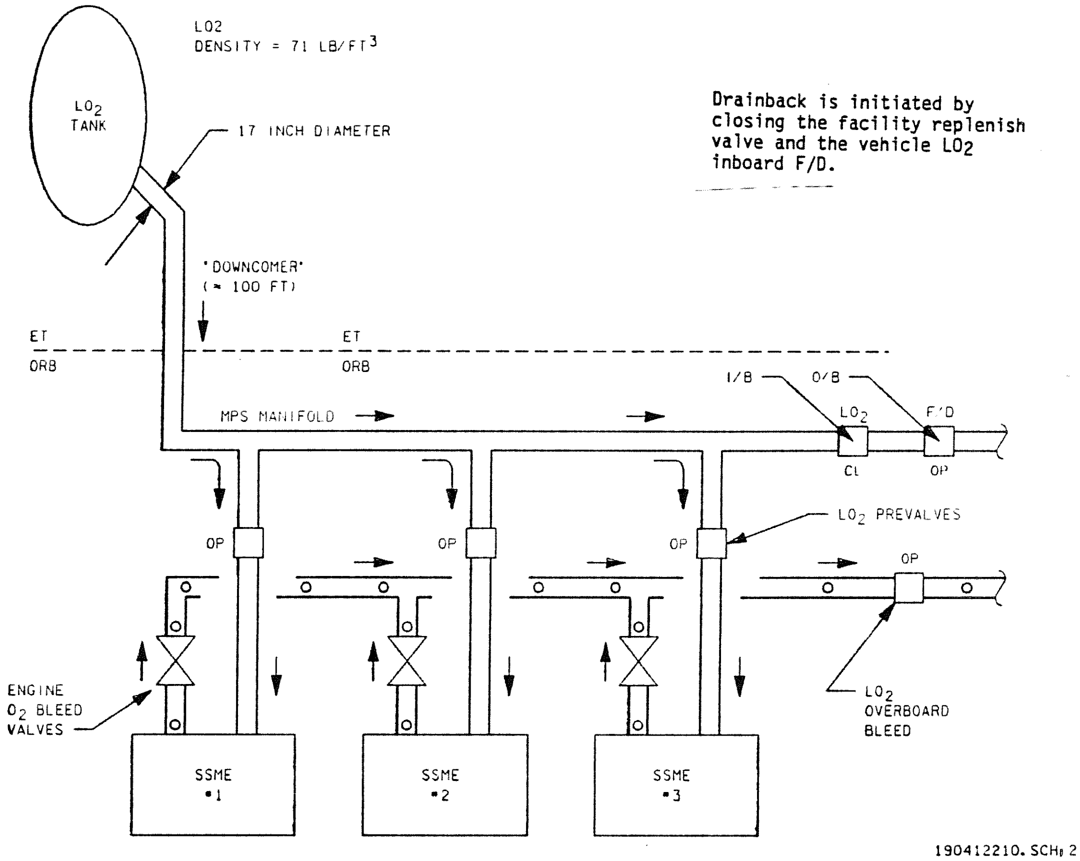
The fluid in the downcomer flows through the engines and out the overboard bleed valve.

The LO2 in the downcomer is not as cold or as dense as the fluid in the tank (due to heat transfer with ambient condition).

Therefore, the engine inlet temperatures will increase to a safe level necessary for engine start.

If a hold is called after drainback initiation, the colder/denser fluid in the ET will eventually sink to the engines. This will cause the temperatures to drop and eventually violate the SSME start box.

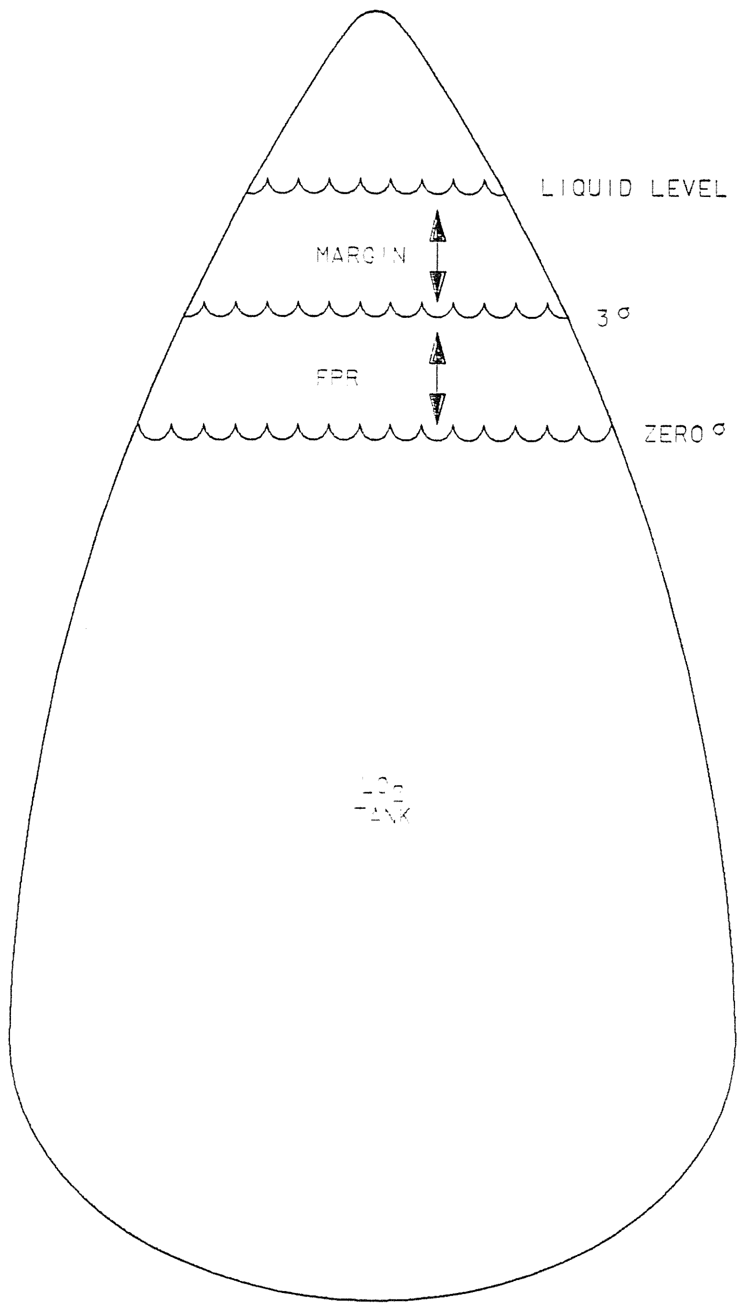
Drainback is initiated by closing the facility replenish valve and the vehicle LO_2 inboard F/D.



LCC: -289.2 F minimum; from T-75 sec to T-31 sec (must be verified prior to restarting count after hold at T-31 seconds)

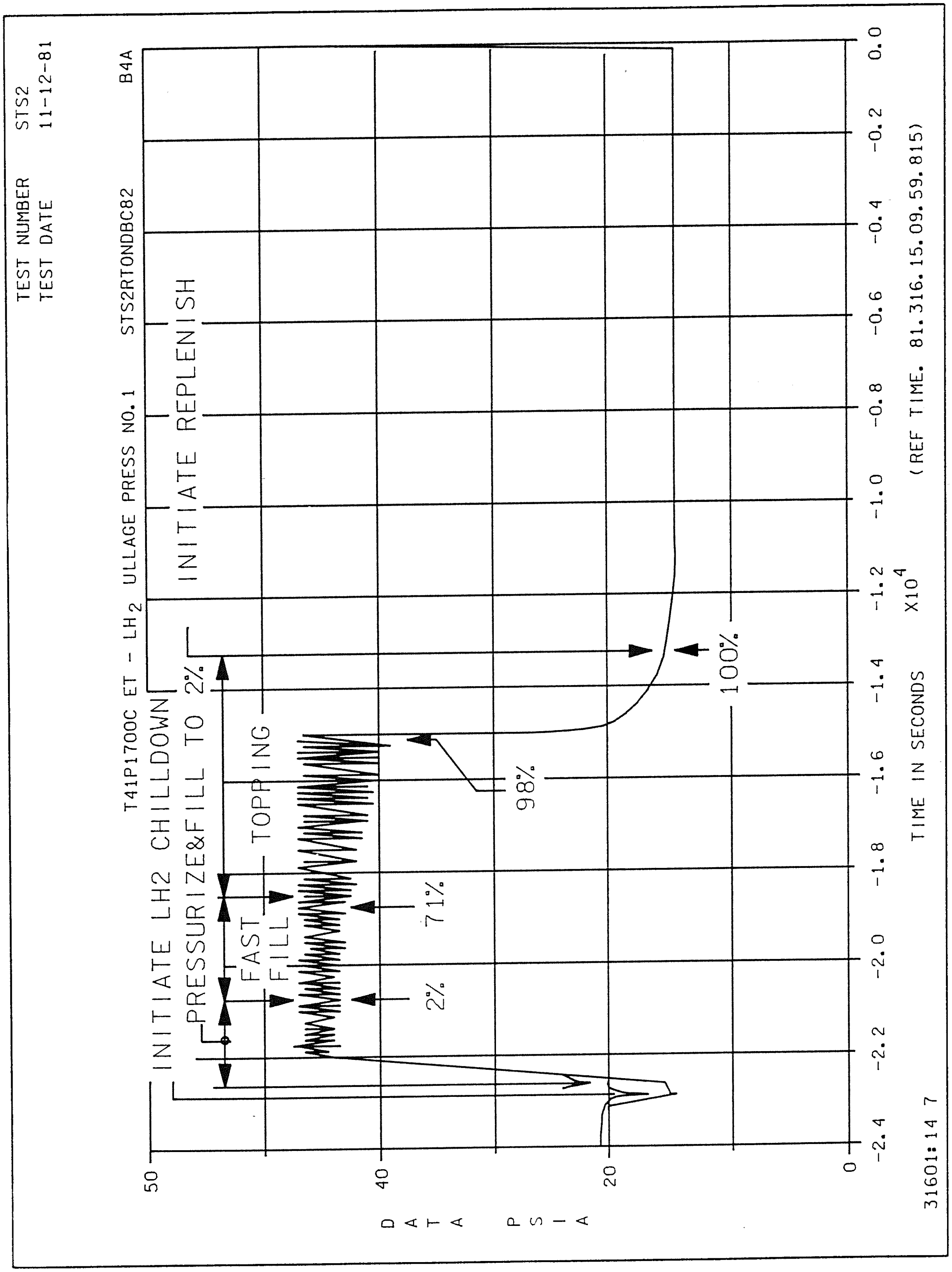
Start box can be violated between 3:30 min to 6:30 min beyond T-0 for drainbacks starting at T-4:55.

Figure 2.2-10.- LO_2 flow during drainback.



190412211.ART;2

Figure 2.2-11.- LO_2 performance drainback margins.



2991.ART,2

Figure 2.2-12.- LH2 ET ullage pressure.

References:

1. OMI No. S0007V2 - Launch Countdown

2. OMI No. G3151 - LO2 System Preps For Vehicle Loading

3. OMI NO. G3251 - LH2 System Preps For Vehicle Loading

4. OMI NO. V9018 - SSME/MPS Preps And Securing For Prop Loading

5. OMI No. S1003 - MPS LO2 System Automatic Load And Drain Operations

6. OMI NO. S1004 - MPS LH2 System Automatic Load And Drain Operations

7. Shuttle Operational Data Book (SODB)

2.3 ET VALVES SCHEMATICS

2.3.1 General

The ET is designed with a minimum of avionics and valve hardware because it is not recovered after launch. The few valves that are part of the ET allow propellant to be transferred from the ground facilities to the ET during preflight and from the ET to the orbiter during powered flight. Relief and vent valves protect the tank from a destructive overpressure, and the tumble valve is used to set the tank into an aerodynamically destructive tumble after separation.

This systems brief describes the valves located on the ET.

2.3.2 Valve Descriptions

2.3.2.1 ET Ground Disconnect

A schematic of the ET ground disconnects (i.e., the GH_2 vent, LO_2 and LH_2 vent valve actuation, intertank purge, hot GN_2 purge, hazardous gas detection, and helium injection disconnects) is shown in figure 2.3-1. These disconnects pass through the ET ground umbilical plate and are spring-loaded closed on the ground side at launch. The GH_2 vent has a 7-inch inside diameter at the umbilical plate.

2.3.2.2 ET Vent And Relief Valves

A schematic of the ET vent and relief valves is provided in figure 2.3-2. The LO_2 and LH_2 vent/relief valves are normally closed, spring-loaded valves that are actuated open by ground support equipment (GSE) helium prior to prepressurization and launch. The LO_2 relief and reseat pressures are 24 ± 1 psig and 22 psig (minimum), respectively. The LH_2 relief and reseat pressures are 39 ± 1 psig and 37 psig (minimum) for the heavyweight tank (HWT) and 36 ± 1 psig and 34 psig (minimum) for the lightweight tank (LWT). The LO_2 valve vents directly overboard, but the LH_2 vent gases are conveyed through the ET/ground carrier umbilical prior to launch.

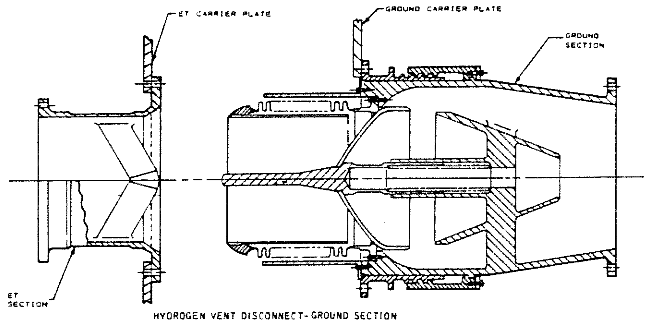
For valve relief mode operations, the two-stage valve design utilizes a primary sensing pilot and a secondary slave pilot. The primary pilot uses local ambient pressure as a reference pressure (sensed at ambient pressure sense port). The primary pilot belleville and bias spring provide control so that valve relief will occur. The secondary pilot allows flow to the main piston in response to a signal from the primary pilot during relief operation. The primary and secondary pilot inlets are connected to the main valve inlet cavity.

The primary pilot is connected to the secondary via drilled passages and is physically bolted to the outside surface of the main body. It utilizes a belleville spring reference vented to atmosphere on one side and ported to valve inlet pressure on the other side. Leakage past the belleville barrier is prevented by a thin stainless steel diaphragm which lays against the belleville face and is welded and clamped to the pilot housing. The pilot poppet is caused to move with the diaphragm belleville by action of a spring which also acts to open the pilot as the belleville moves outward due to increasing tank pressure thereby porting tank gas pressure to the secondary pilot.

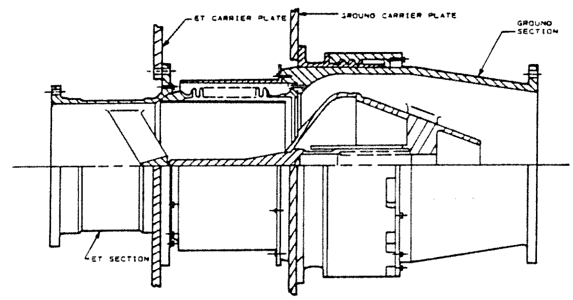
Secondary pilot inlet pressure is admitted to both sides of an actuating piston which is directly coupled to the pilot poppet. The downstream side of the actuating piston is ported to the primary pilot poppet. When the primary pilot opens, flow past the restrictor metering pin causes pressure drop across the secondary actuating piston head which unseats the secondary pilot poppet allowing flow to the main valve actuator piston. This, in turn, forces the main poppet open thereby relieving the tank. The secondary pilot incorporates a balancing piston, the downstream side of which is vented to a low pressure area. This prevents instability which would be caused by pressure fluctuation in the main valve actuator section.

2.3.2.3 ET Orbiter Disconnects

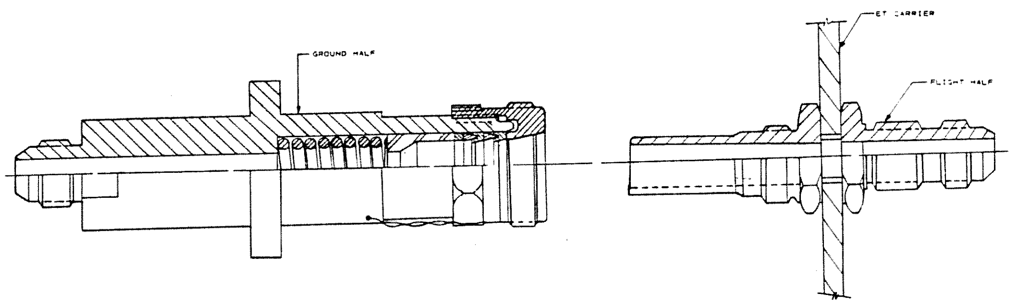
A schematic of the ET orbiter disconnects (i.e., the LO_2 and LH_2 feedline, LO_2 and LH_2 tank pressurization, and LH_2 recirculation disconnects) is shown in figure 2.3-3. The feedline, pressurization, and recirculation disconnects are 17 inches, 2 inches and 4 inches in diameter, respectively, and pass through the LH_2 and LO_2 ET/orbiter umbilical plates. Each feedline disconnect consists of a pair of two-position, pneumatically-actuated, flapper valves (one on each side of interface). In the open position, an open direction force is maintained on the rotary flapper valves by a locked, over-center mechanism. In addition to the linkage force, the flappers are positioned in the flow stream such that the fluid force increases the open force as the flow increases. The design of the flapper linkage provides for relief of pressure buildup between the two flappers when the units are mated by allowing the flapper to lift off the seat against a spring force. Before umbilical separation, these valves are pneumatically closed by orbiter actuators. Linkages across the interface close the tank half of each disconnect. These valves are also mechanically driven closed when the umbilical plates are retracted. On STS-26 and Subs, a pnuematically actuated latch will be added to the disconnect valves to avoid possible valve closure during SSME operation. The disconnect valves will be latched open prelaunch and the latch mechanisms will be disengaged after MECO but prior to ET separation. The LH_2 recirculation disconnect, with one closure device on each side of interface, is pneumatically held open during boost. Before separation, the valve closure devices are pneumatically closed. As a backup, the disconnects will be spring-loaded closed if pneumatics is lost. The recirculation and pressurization disconnects are mechanically pushed to the open position by engagement of the two closure devices (one on each side of interface) and are spring-loaded closed when the umbilical plates are retracted.



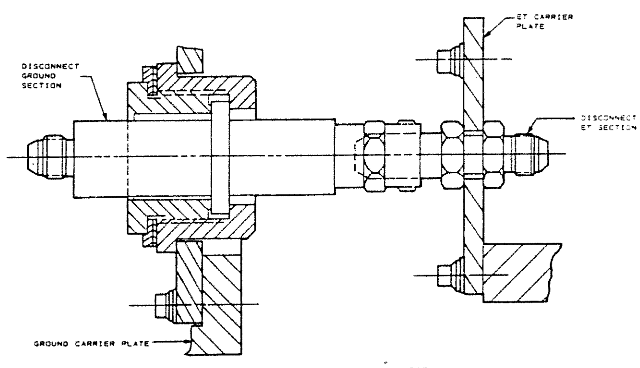
(a) Hydrogen vent disconnect - not mated.



(b) Hydrogen vent disconnect - mated.



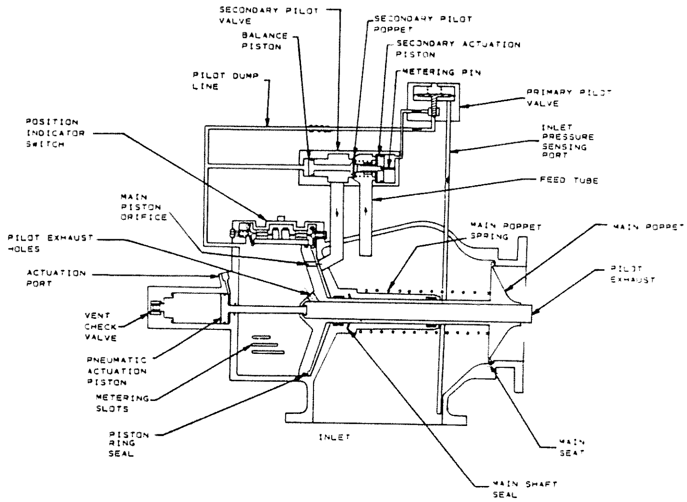
(c) Pressurization disconnect - not mated.



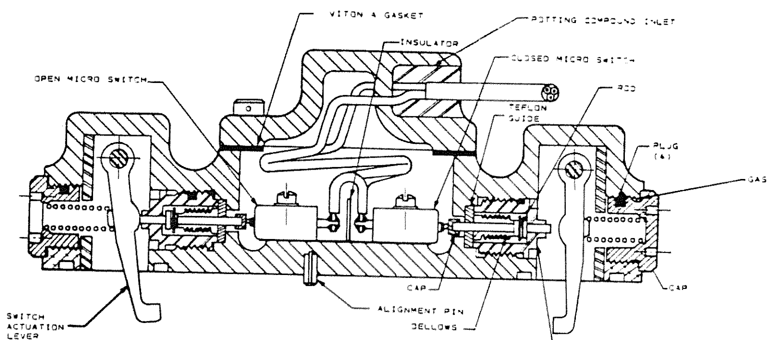
DASH NO.	USE
-010	1. LO_2 TANK VENT VALVE ACTUATION
-009	2. LH_2 TANK VENT VALVE ACTUATION
	3. INTERTANK PURGE
	4. HOT GN_2 PURGE (NOSE CONE)
	5. RSS EQUIPMENT PURGE
	6. HAZARDOUS GAS DETECTION SYS.
	7. ELECTRIC HEATER PURGE

(d) Pressurization disconnect mated.

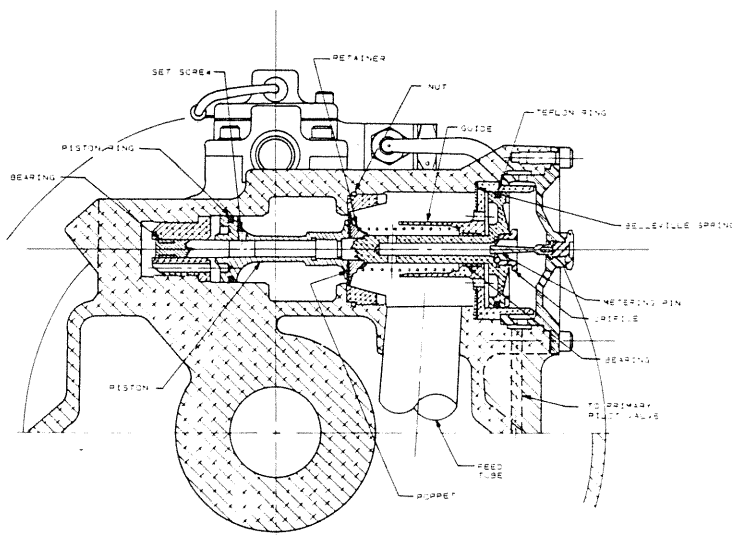
Figure 2.3-1.- ET ground disconnects.



SCHEMATIC DIAGRAM



SECTION A-A
MICRO SWITCH ASSEMBLY



SEC ION B-B
(SECONDARY PILOT)

Figure 2.3-2.- ET vent and relief valves (page 1 of 2).

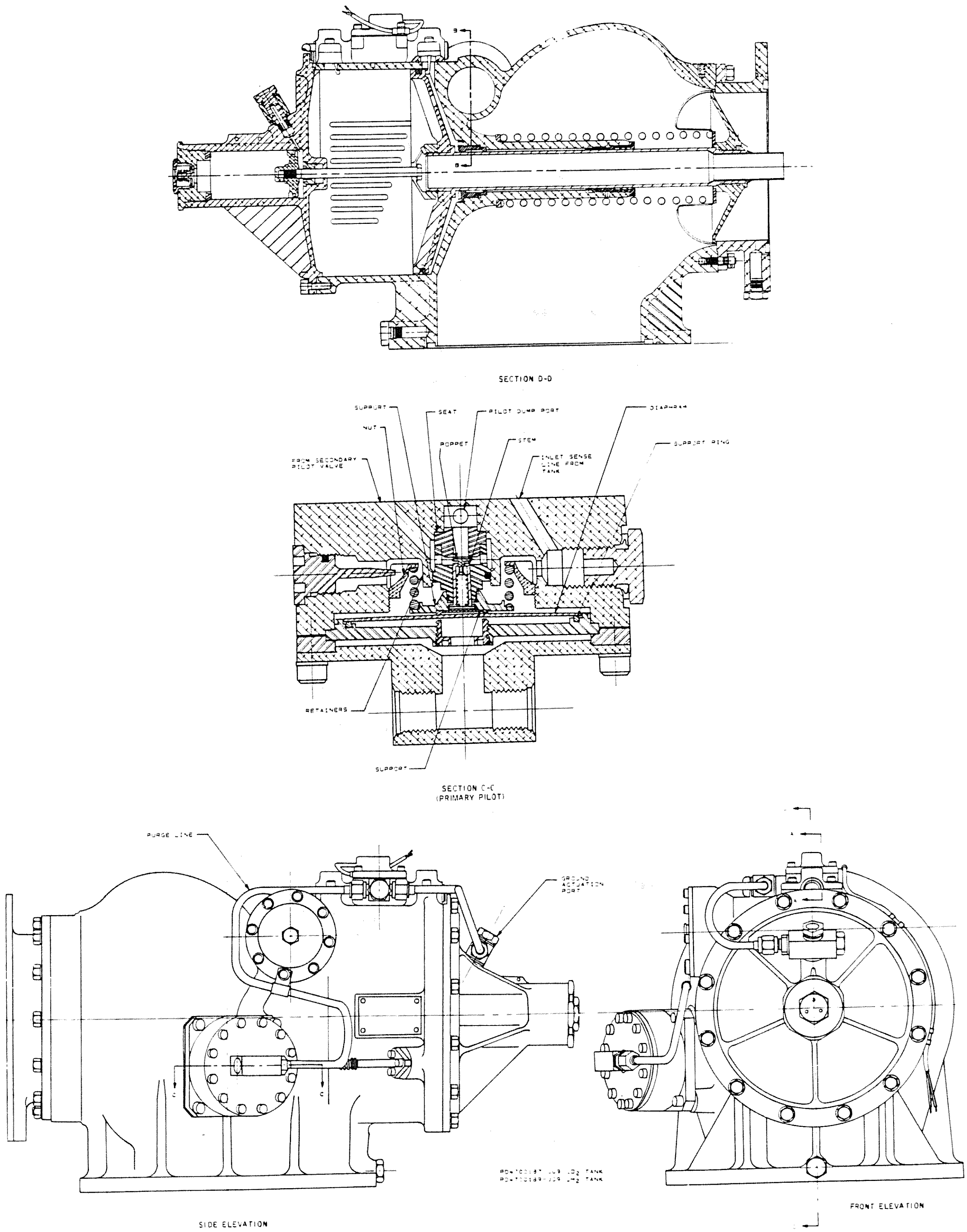
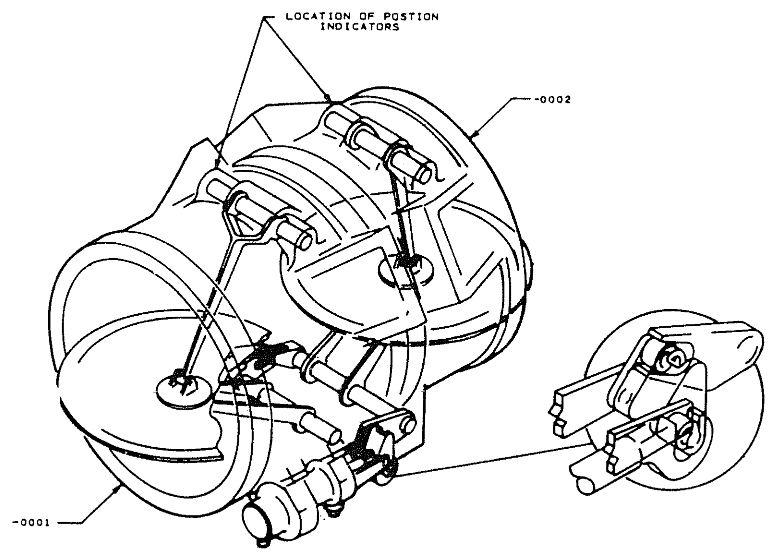


Figure 2.3-2.- ET vent and relief valves (page 2 of 2).

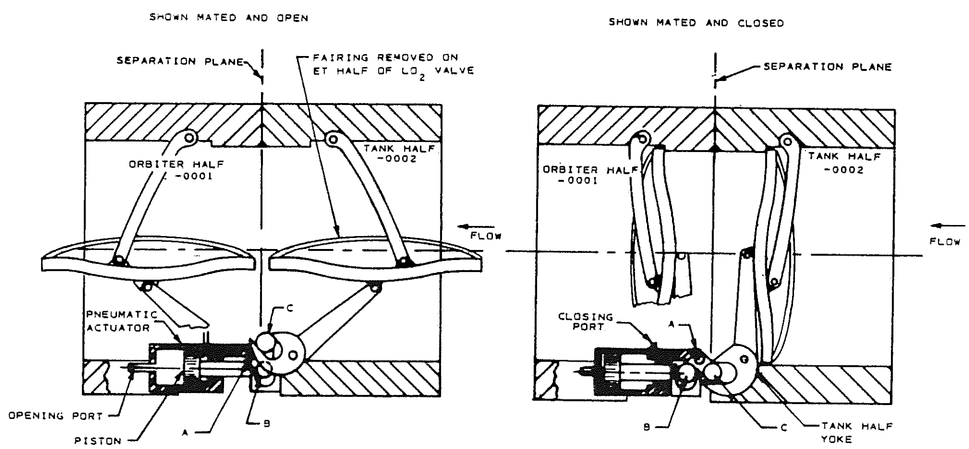


17 INCH FEEDLINE DISCONNECT

ORBITER HALF
MC284-0389-0001

ET HALF
MC284-0389-0002

DASH NO	USE
0001	1. LO_2 & LH_2 ORB HALF 17 IN FEEDLINE DISC VALVE
0002	2. LO_2 & LH_2 ET HALF 17 IN FEEDLINE DISC VALVE



1. PRESSURE APPLIED TO OPENING PORT MOVES PISTON TO RIGHT ROTATING BELL CRANK ABOUT POINT A. POINTS B & C ARE PERPENDICULAR TO PISTON CENTER LINE. (PRELAUNCH)

2. PRESSURE APPLIED TO CLOSING PORT MOVES PISTON TO LEFT ROTATING BELL CRANK ABOUT POINT A. POINTS B & C ARE IN LINE WTIH PISTON. (POST MECO)

3. WHEN UMBILICAL PLATE IS RETRACTED PRIOR TO ET SEPARATION, BELL CRANK SLIDES AWAY FROM TANK HALF YOKE. (ET SEPARATION)

4. PLATE RETRACTION CAUSES ROTATION OF BELL CRANK AS IN STEP TWO PROVIDING A BACKUP TO NORMAL PNEUMATIC CLOSURE

Figure 2.3-3.- ET orbiter disconnects (page 1 of 3).

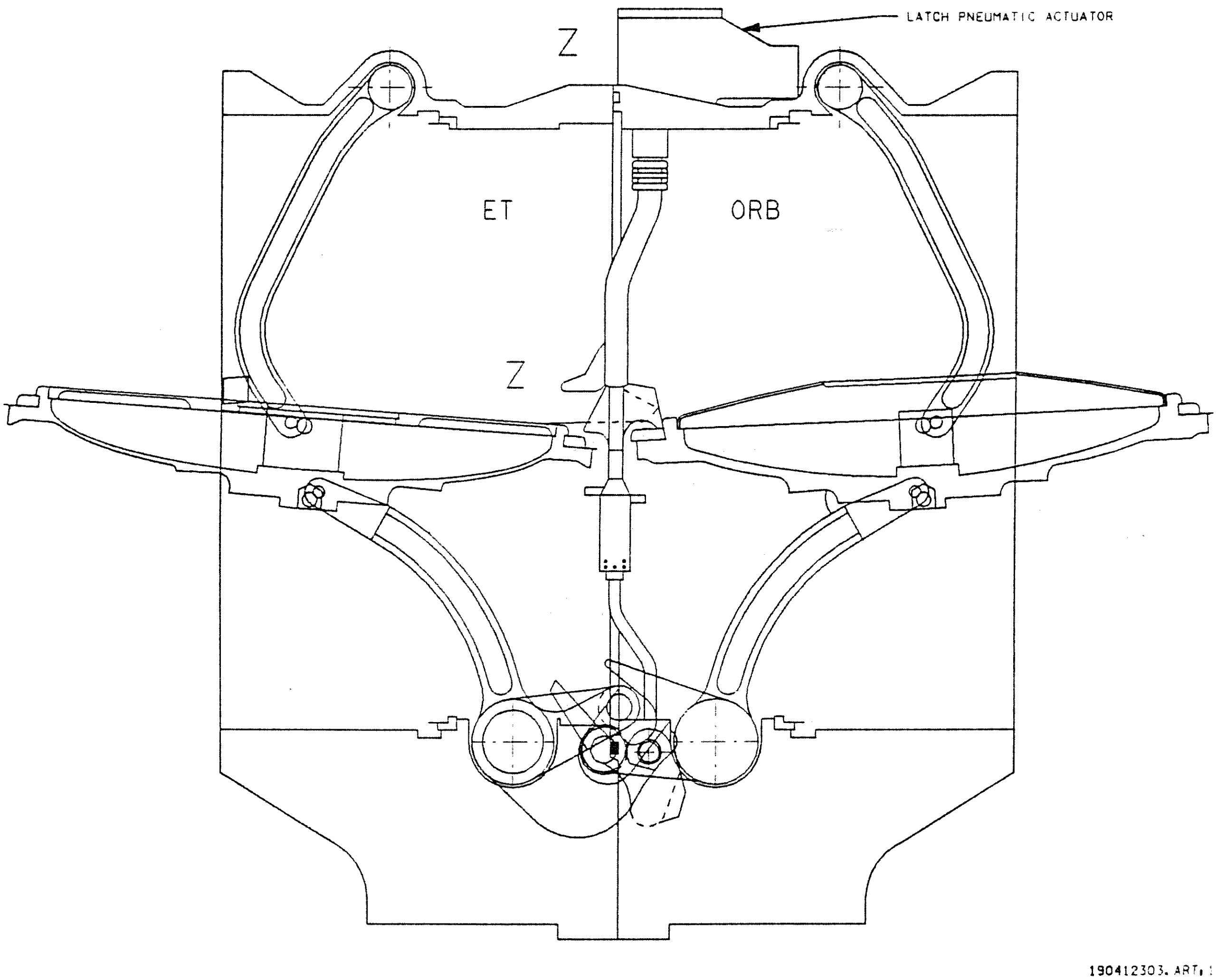
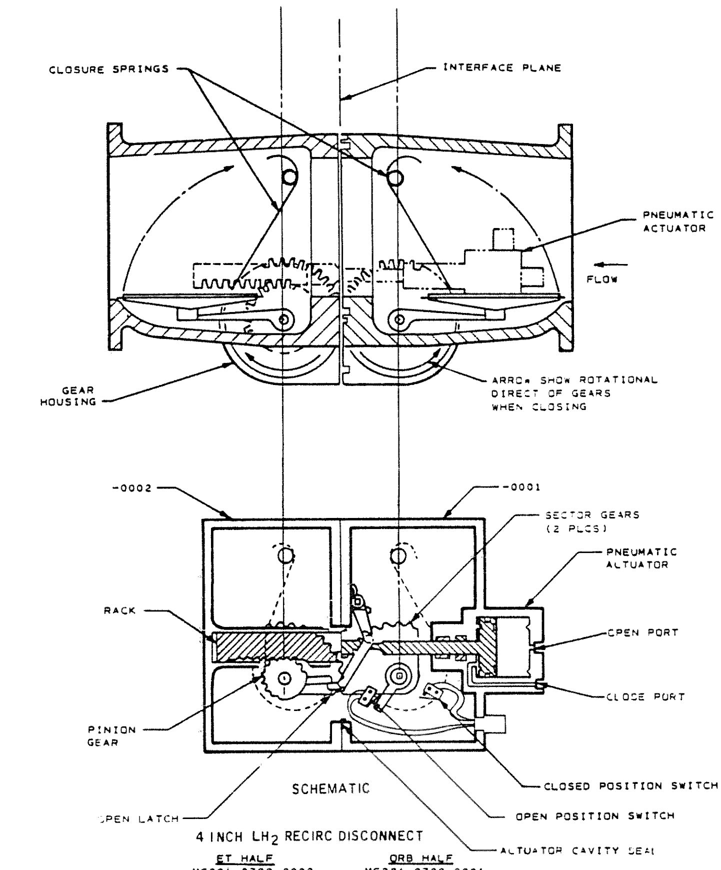


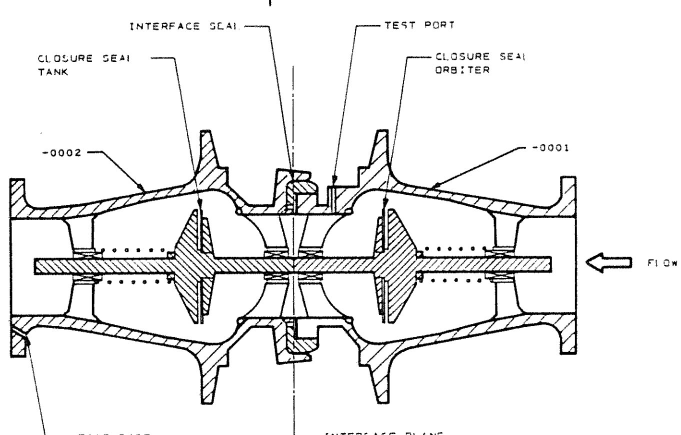
Figure 2.3-3.- ET orbiter disconnects (page 2 of 3).



4 INCH LH_2 RECIRC DISCONNECT

ET HALF	ORB HALF
MC284-0390-0002	MC284-0390-0001

DASH NO	USE
0001 & 0002	1. LH_2 ET/ORB RECIRCULATION DISCONNECT VALVE



ET / ORB PRESSURIZATION DISCONNECT

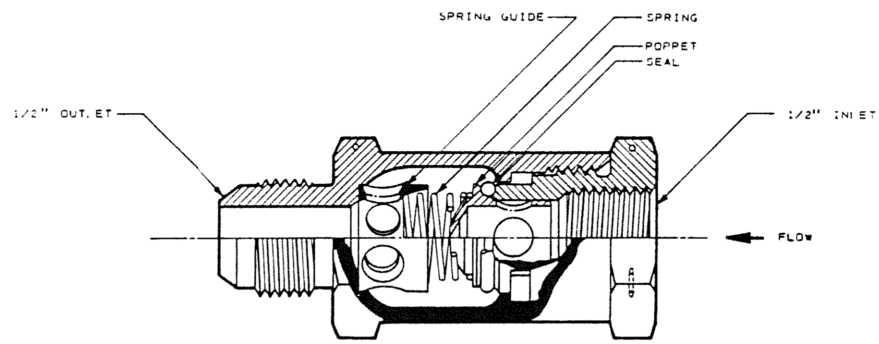
ET HALF	ORB HALF
MC284-0391-0002	MC284-0391-0001

DASH NO	USE
0001	1. ET LO_2 TANK PRESSURIZATION DISC.
0002	2. ET LH_2 TANK PRESSURIZATION DISC.

Figure 2.3-3.- ET orbiter disconnects (page 3 of 3).

2.3.2.4 ET Check Valves

A schematic of the ET check valves is given in figure 2.3-4. The ET check valves (two in series) provide flow isolation for the helium injection system line.



MARTIN MARIETTA STD PART

PART NO	USED
4711	HELIUM INJECT SYSTEM CK VLV (2 REQUIRED - IN SERIES)

Figure 2.3-4.- ET check valve.

2.3.2.5 ET Tumble Valve

A schematic of the ET tumble valve is shown in figure 2.3-5. The 2-inch, pyro tumble valve mounts to the top of the LO2 tank. A NASA standard initiator (NSI), together with a pyro booster, generates the explosive pressure that causes a piston/ram to shear a gate within the pyro valve. This allows LO2 tank ullage gas to flow through the valve and exit the ET through an external nozzle. The purpose of the valve is to provide positive tumbling of the ET post-separation to reduce the ET impact point footprint.

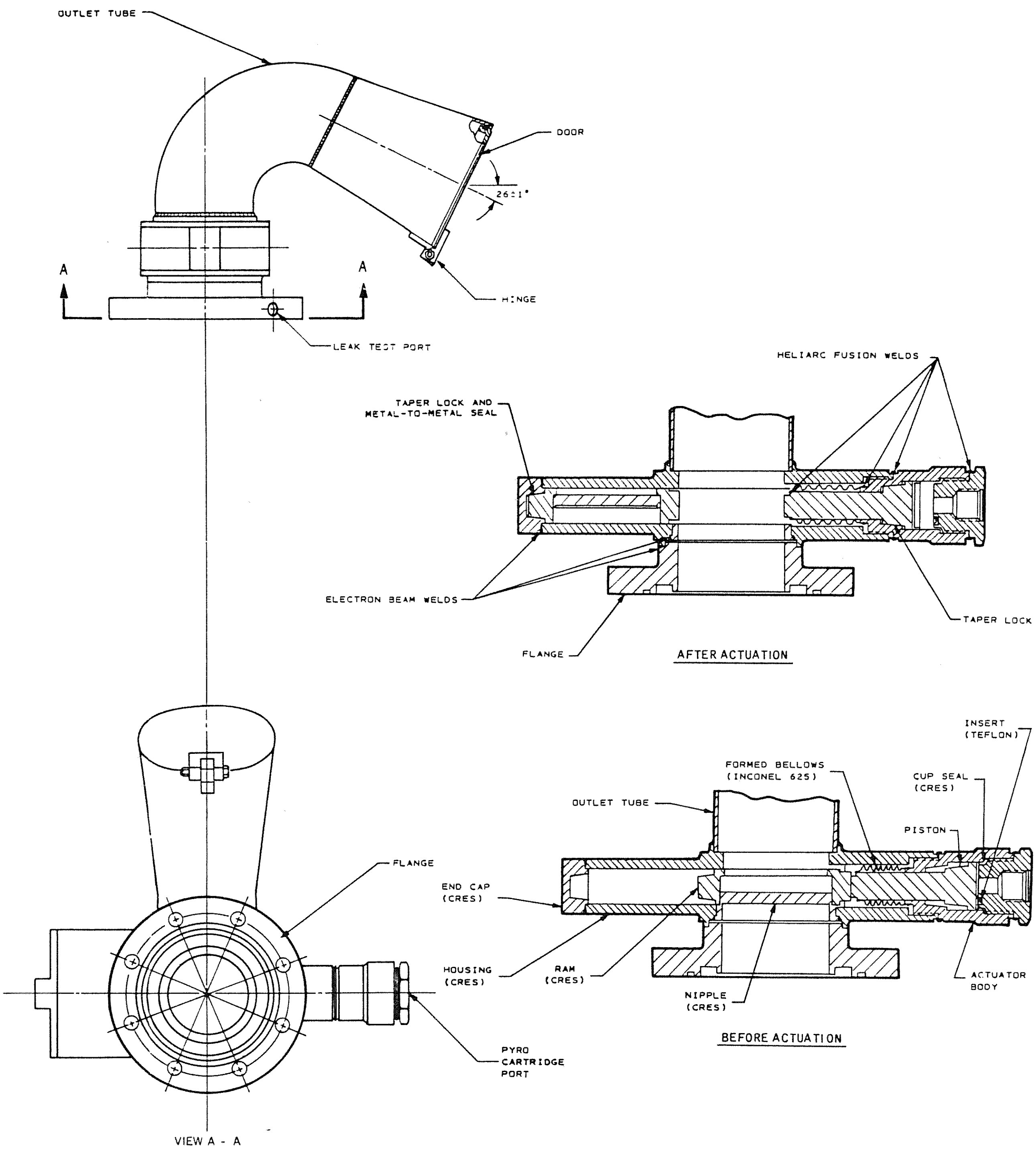


Figure 2.3-5.- ET tumble valve.

2.4 ET LH_2/LO_2 LOW LEVEL CUTOFF BRIEF

2.4.1 General

Nominal MECO is initiated when the vehicle reaches a desired I-loaded velocity. If an off-nominal condition causes the propellant supply to be depleted before the MECO velocity is reached, then the ET LH_2/LO_2 low level cutoff function will shut down the SSME's before the propellant supply (LO_2 or LH_2) to the engines is depleted. Depletion of LO_2 to a running engine will cause pump cavitation and possible uncontained damage to engine components. With LH_2 depletion, an LO_2-rich shutdown of a running engine will occur. This LO_2-rich shutdown causes higher than normal temperatures and results in burning and severe erosion of engine components. To ensure that all depletion cutoffs are not LO_2 rich, an approximate 1000 lbm bias is included in the amount of LH_2 loaded in the ET. This amount exceeds that dictated by the 6:1 (oxidizer lbm/fuel lbm) engine mixture ratio.

2.4.2 Functional Description

Four LO_2 low level sensors are located in the orbiter 17-inch LO_2 feedline and four LH_2 low level sensors are located in the ET LH_2 tank. These sensors are positioned to ensure sufficient propellant between the sensors and the SSME pump inlets to prevent propellant depletion during engine shutdown. To prevent a premature SSME shutdown caused by three failed-dry sensors during flight, the low level sensors are not armed until an arm command is received from guidance.

The ET LOW LEVEL SENSOR ARM CMD is initiated by the ascent second stage guidance function or the powered RTLS guidance function. From either source, the ARM command is sent to the SSME's under the following conditions:

- A second SSME has failed

- The current calculated vehicle mass (M) becomes less than an I-loaded value (MASS LOW LEVEL V97U4432C)

This I-loaded value is selected to provide the orbiter with a TAL underspeed capability should MECO occur when the ARM command is received. This would happen if three LO_2 (or LH_2) level sensors had failed dry after lift-off.

During flight, when the ET LOW LEVEL SENSOR ARM CMD is received, the SSME operations sequence monitors the eight low level sensors for dry indications or commfault indications. Sensors that have been commfaulted are disregarded. On the first pass through the logic after the arm command if any sensors indicate dry the logic will disable up to one LO_2 and one LH_2 sensor. This check allows for the removal from the logic of a failed dry sensor.

On subsequent passes through the logic if two or more of the LH_2 sensors (not previously disabled or commfaulted) indicate dry, the sequence issues a

MECO command and the SSME's are shut down. If an LO_2 sensor (not previously disabled or commfaulted) indicates dry, an internal flag is set, indicating that sensor is dry. When any two LO_2 sensor dry flags are set, one of two shutdown timers is started.

Timers are used with the LO_2 low level sensors to delay the MECO command after two sensors are dry. This delay ensures maximum allowable consumption of LO_2 during engine shutdown while still preventing LO_2 depletion. The time between two sensors dry and LO_2 depletion at the SSME's is dependent upon an LO_2 flow rate proportional to throttle setting. The time delay (timer) used for each mission leg is shown in table 2.4-I. Rationale for the timer selection follows.

TABLE 2.4-I.- RATIONALE FOR TIMER SELECTION

MSID	Mission leg	Shutdown power level (%)	Timer* (sec)
V97U9864C	RTLS	65/104	0.0
V97U9863C	Nominal	65	0.398
V97U9865C	AOA	91	0.0

*Timer values may vary from flight to flight. Refer to PASS I-Loads for specific values.

For a two-engine RTLS with one stuck at 104 percent and one throttled to 65 percent before MECO, total throttle is 169 percent. Although the total throttle is less than a nominal MECO, the time delay will be shorter because the shutdown of an engine from 104 percent requires a higher NPSP. The shorter 0.0-second timer will provide a larger residual of LO_2, but the NPSP requirement will not change. The NPSP requirement for MECO is driven by the engine at the highest power level. Furthermore, the actual NPSP delivered at MECO is not solely a function of residuals, but is strongly dependent on tank pressure at g-level. All two-engine RTLS shutdowns will use the 0.0-second timer.

For the nominal case, when all three engines shut down from 65 percent, the 0.398-second timer is used. Although the total throttle for the nominal MECO (195 percent) is greater than the RTLS case (169 percent), the shutdown of all engines from 65 percent requires a lower NPSP and therefore a lower head of LO_2.

The two-engine AOA where the engines are throttled to 104 percent and both are shut down from 91 percent for a total of 182 percent will use a timer setting of 0.0 seconds. The total throttle percentage is close in all three situations and the determining factor for the timer value is the NPSP required at shutdown. After the shutdown timer times out, a MECO command is issued and the SSME's are shut down. Such a shutdown is known as a low LOX cutoff.

No timers are used with the LH_2 low level sensors. Since the loaded LH_2 is biased by approximately 1000 lbm, it would take one failure beyond a LO_2 low level depletion to cause a LH_2 depletion. The LH_2 low level sensors are positioned to ensure optimum conditions for a nominal three engines at 65 percent shutdown.

MECO initiated by low level sensors is a flight anomaly that generally produces a flight performance penalty. To reduce that penalty, the SSME's must be allowed to run as long as possible without depleting either propellant. The low level sensor shutdown logic covered in this SB provides the boundary between propellant depletion and excessively early engine shutdown.

The eight ET low level sensors are connected to one point sensor electronic package located on the orbiter. The electronic package is energized using 28 V dc power during checkout, propellant loading, and the ascent portion of flight. The system is deenergized within 5 minutes after main engine shutdown.

The transducers shown in figure 2.4-1 use a thermosensor principle to detect liquid/gas. A constant current is fed to the transducers and the resultant voltage is monitored to detect the fluid condition. Voltage is low when the transducer is immersed in a liquid, since heat transfer is high and the platinum sensing element temperature stays low keeping the element resistance low. The effect occurs when the transducer is in a gas. The electronics box powers and monitors the sensor and issues wet (zero volts) and dry (28 volts) signals. If the platinum sensing element should break, creating an open circuit, the electronics box detects a no-current flow condition and issues a wet signal. If the resistance increases the wiring between the electronic package and a sensor element, it is possible to have a false dry indication. This resistance must be between 11.3 and 500 ohms for the sensor to indicate dry.

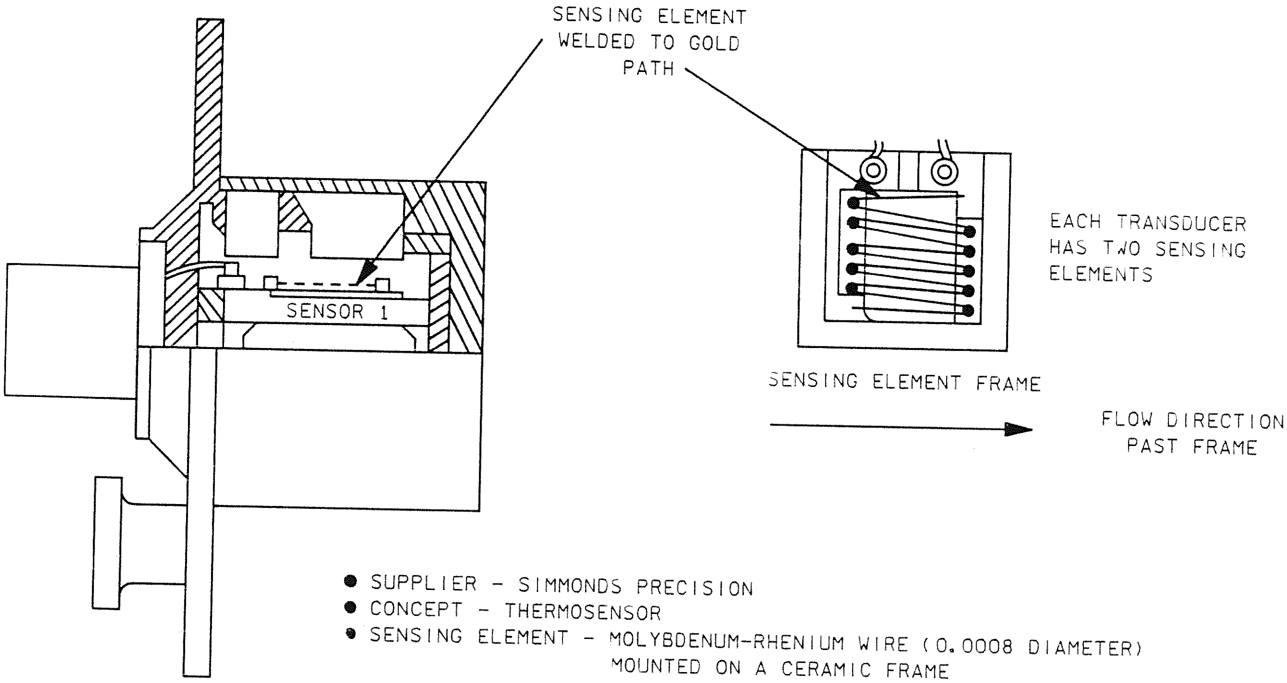


Figure 2.4-1.- Orbiter LO_2 MECO point sensor.

2.5 EXTERNAL TANK SEPARATION

2.5.1 Introduction

External tank (ET) separation is an important part of the ascent profile. It occurs approximately 8.5 minutes after liftoff, immediately following main engine cutoff (MECO). Although the entire ET separation process normally requires less than 30 seconds, a complex interaction of hardware and software is required for its successful completion.

The ET supplies LH_2 and LO_2 propellants to the three SSME's. It also serves as the structural backbone for the orbiter and the SRB's during ascent. At 154 feet long and 27.6 feet in diameter, it is the largest component of the space shuttle vehicle. Empty, it weighs approximately 69,000 lb. When fully loaded with propellant at launch, it weighs approximately 1,660,000 lb.

External tank separation is normally an automatic process. However, the crew has the capability to manually override the auto separation sequence or to manually initiate it any time in OPS 104, 105, 106, or 601. A contingency separation sequence called "fast sep" can be initiated manually by the crew in OPS 102, 103, or 601. After separation, the tank pitches away from the orbiter, tumbles, and breaks up upon reentry. Any pieces of the tank that survive entry fall within a designated ocean impact area.

This systems brief discusses the hardware and software (PASS, Flight Software Version OI-21) involved in the ET separation process. Attention focuses primarily on nominal separation, although abort and off-nominal cases are addressed. It also describes crew interfaces with the separation process and explains how manual separation is accomplished.

2.5.2 Crew Interfaces

The crew can interface with the ET separation process via several switches and SPEC item entries. These interfaces allow the crew to control the activities of the automatic process.

2.5.2.1 ET Separation AUTO/MAN Switch and SEP Pushbutton

These switches are illustrated in SSSH drawing 10.1, sheet 2 of 4. They provide a means for the crew to manually override the automatic separation process and initiate physical separation of the orbiter and external tank. Separation is normally accomplished without crew intervention with the mode switch in AUTO. If there is reason to delay ET separation or override the automatic separation sequence, the crew can place the mode switch in the MAN position. This inhibits physical separation until the ET SEP pushbutton is depressed or the switch is returned to AUTO.

Each switch position has three contacts. Two of three voting logic determines the position of the switch. Default logic is provided to resolve dilemmas. If failures cause the switch to indicate both the AUTO and MAN position simultaneously or cause no position to be indicated, a dilemma is declared. In OPS 1, switch RM defaults to the MAN switch position. An "ET SEP MAN" message is annunciated to inform the crew. In OPS 6, switch RM defaults to the AUTO position and an "ET SEP AUTO" message is annunciated.

The override display, SPEC 51, is available to the crew and allows manual override of the switch in the presence of switch failures.

2.5.2.2 Override Display - Spec 51, item 38 and item 39 (BFS 28, 29)

The PASS override display, SPEC 51, is illustrated in figure 2.5-1. Item entries 38(28) and 39(29) are legal in MM102, 103, 104, 105, 106, and 601. Item 38(28) provides the capability to override a manual ET separation and reestablish AUTO. It is used when switch failures cause a manual default to be selected. Item 39(29) is used when automatic separation is inhibited and the ET SEP AUTO/MAN switch or SEP pushbutton are inoperative. This Spec option initiates physical separation of the orbiter and ET.

2.5.2.3 ET Umbilical Door Switches

These switches are located on panel R2 and shown in SSSH drawing 15.2, sheet 2 of 4. They are used by the crew after ET separation to stow the centerline latches, close the ET umbilical doors, and latch the doors. Umbilical door hardware is discussed in greater detail in a later section.

2.5.2.4 Override Display - Spec 51, item 40 (BFS 30)

This item entry is illustrated on the PASS override display, figure 2.5-1. It serves as a backup to the ET umbilical door switch. Item 40 is only legal in MM104, 105, and 106.

2.5.3 ET Sep Hardware

To understand the steps executed by the software during ET separation, it is helpful to understand the hardware that is involved in the process. A brief description of this hardware follows.

2.5.3.1 Pyro Initiator Controller

Pyro initiator controllers (PIC's) are explosive devices used throughout the separation process to sever nut/bolt assemblies mating the orbiter and ET. They have a standard design that is used throughout the orbiter vehicle. PIC's remain unarmed until they receive an arm command, which consists of a 20-32 V dc signal. This voltage is actually input power for a dc/dc

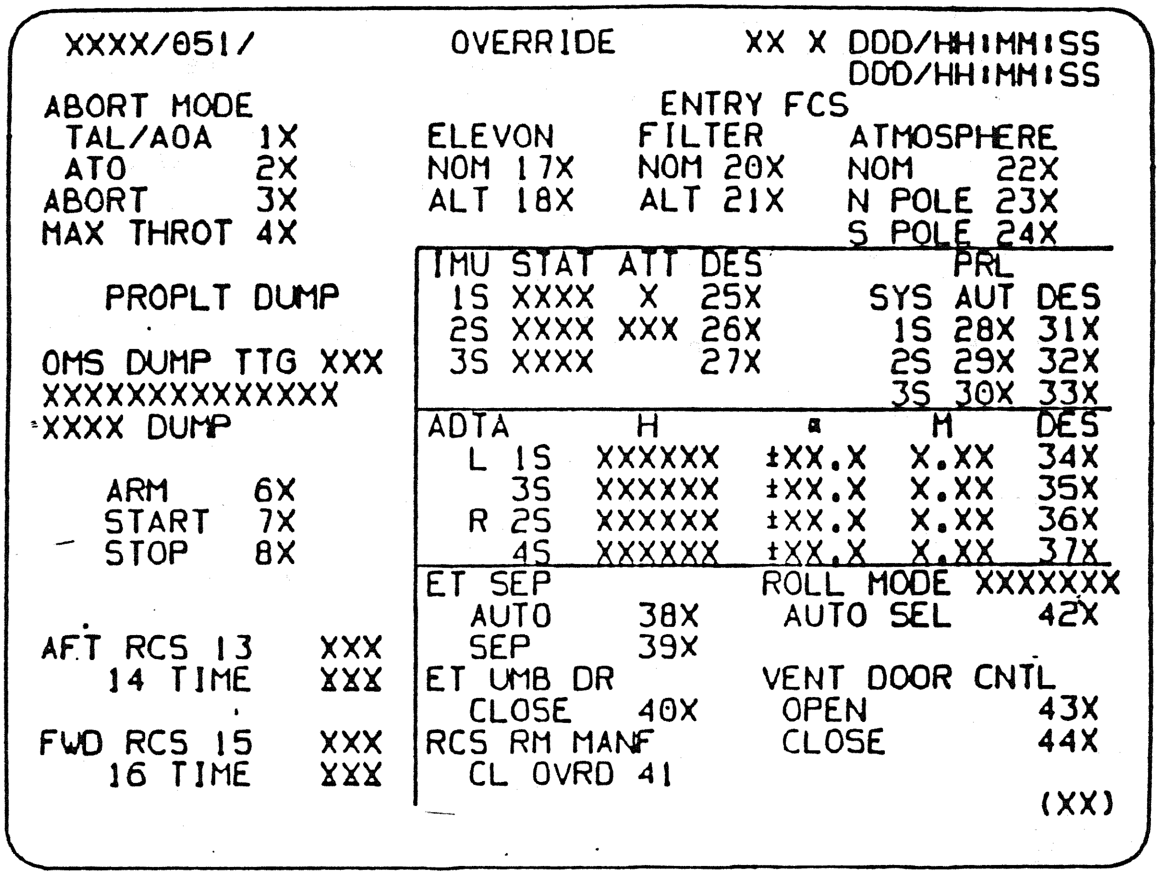


Figure 2.5-1.- Override display.

converter. The dc/dc converter charges a capacitor bank which stores sufficient energy to fire a NASA standard initiator (NSI). Once armed, a FIRE command causes the capacitors to discharge to the NSI, creating an explosion.

2.5.3.2 Forward and Aft ET Structural Attach Points

There are three orbiter/ET attach points, one forward and two aft. The forward attach point is supported from the LH_2 tank forward ring frame. The aft structure is mounted on the ET aft major ring frame and the LH_2 tank longerons. Refer to SSSH drawing 15.3, sheet 1 of 3. Numerous pinned and spherical joints exist within the support structures that allow for multidirectional motion, minimizing bending induced in the support structures as a result of thermal and structural loads.

2.5.3.2.1 Forward structural attach point.- The forward attach point is designed with rotational freedom about a Y-axis reference line so that changes in ET length resulting from thermal effects will not induce loads into the orbiter. ET attach hardware consists of two struts joined to a yoke fitting. A frangible bolt runs through the apex of the yoke and is secured inside the orbiter with a large hex nut. Refer to SSSH drawing 15.3, sheet 1 of 3. This nut is equipped with two pyros, referred to as the structural separation PIC's, which explode at ET structural separation and shear the nut. Either of the two structural separation PIC's is capable of fracturing the nut. A containment vessel captures the exploded nut fragments. The strain energy of the bolt, which is released when the nut is exploded, causes the bolt to spring out of its yoke and travel away from the orbiter. This motion is aided by Bellville springs, which push the freed bolt outside the orbiter along a guide rod. A device called a hole plugger trails behind the exiting bolt and fills the vacated area. No closeout door is required to cover the forward attach point after separation.

2.5.3.2.2 Aft structural attach points.- The two aft ET structural support assemblies are slightly different from one another. The starboard aft support is a tripod, making it fixed, while the port structure is a bipod that permits lateral pivotal motion. This design allows unrestrained relative lateral distortion between the aft ET and the orbiter. A rectangular crossbeam connects the two support structures.

The ET/orbiter structural interface attachment hardware is the same at both aft attach points. It is very similar to the hardware in the forward assembly. A 2.5-inch diameter tension bolt extends from the apex of the bipod (tripod) assembly on the ET and engages a frangible nut housed in the orbiter. Each nut is equipped with two structural separation PIC's. ET/orbiter physical separation is accomplished by firing these PIC's which shear the nut. Each bolt has a retracting spring on the ET, which retracts the bolt into the ET strut after nut fragmentation. A spring-loaded stopper on the orbiter assists in forcing the bolt away from the orbiter and fills the cavity vacated as it exits. Fragments of the exploded nut are confined within a blast container on the orbiter.

2.5.3.2.3 ET Umbilical Attach Plate.- The ET/orbiter fluid and electrical interfaces are located at two aft umbilical plate assemblies. These plates are adjacent to, but separated from, the two aft structural attach points. Orbiter/ET interfaces located on the umbilical plates include a 2-inch tank pressurization line, a 17-inch feedline, a 4-inch tank recirculation line (LH_2 side only), and an electrical monobal disconnect. In the special case of the OV-102 vehicle, separation cameras are also installed on the umbilical plate and provide coverage of the ET/orbiter separation process. Refer to SSSH drawing 15.3, sheet 2 of 3.

Each orbiter umbilical plate is mated with its counterpart on the ET via three frangible nut/bolt combinations. Each nut is equipped with two pyrotechnic detonators. During the ET/orbiter separation process these pyros, referred to as the ET umbilical unlatch PIC's, are fired. This shears the nuts, allowing the mated umbilical plates to separate.

Each umbilical plate is equipped with three hydraulic retractors. These retractors pull the plate approximately 2.5 inches into the orbiter following release of the three frangible nut/bolt combinations. A solenoid valve incorporated in the retractor device controls fluid flow to the retractor. The solenoid valve is enabled by ET umbilical retract PIC's.

The use of the word PIC in the name of the ET umbilical retract PIC's is somewhat misleading. This is not an explosive device because it does not contain an NSI. The electrical charge generated by the PIC is used instead to supply electrical power to the solenoid valve. When the solenoid valves are powered, they allow hydraulic fluid to flow to the hydraulic retractors. Only two of the three hydraulic actuators are required to retract the umbilical plates successfully.

Retracting the umbilical plate serves three functions. Primarily, the retract motion disconnects the orbiter/ET electrical umbilical in the first one-half inch of travel. This releases fluids trapped between the orbiter and ET LO_2/LH_2 shutoff valves. A secondary function of the retractor motion is to provide a redundant means for closing the 17-inch LO_2 and LH_2 disconnect flapper valves whose primary actuation force is pneumatic. As the umbilical plate retracts, these flapper valves are forced closed via mechanical linkages. This function is especially important because it is the primary, and only, means of closing the 17-inch disconnect valves during a fast ET separation (to be discussed later). In addition, each orbiter umbilical plate has three X-Y plane stabilizing bungees that hold the plate in position after separation from the ET umbilical plate.

2.5.3.3 ET Umbilical Doors

After the ET separates from the orbiter, the two aft umbilical openings are exposed. Two doors with thermal protective tiles cover the exposed umbilical areas shielding them from entry heating. These doors are approximately 50 inches square. Each door is driven open/closed by two, three-phase ac motors and hinged to the orbiter by two hinges. The doors also have two different types of latches: centerline latches hold the doors open during ascent, and uplock latches hold the doors closed during orbit and entry. Refer to SSSH drawing 15.2, sheet 1 of 5.

The doors are held in their ascent stow position by two centerline latches that engage both the left and right doors. Each centerline latch engages a fitting on the outer edge of the doors. After ET separation, the latches are disengaged and allow the doors to close. This is accomplished by rotating the latches 58° (a minimum rotation of 33.75° is required to free the doors). After rotating, the two centerline latches retract into grooves etched in the belly of the orbiter and are then considered stowed.

After they are released by the centerline latches, the umbilical doors are closed and then held in place by uplock latches. There are three uplock latches located inside each umbilical cavity. The latches engage three uplock rollers located on the inside of each door. The latches are designed to prevent the doors from vibrating or reopening during entry. When the door is within 2 inches of being closed, the three uplock rollers contact the three uplock latches. Ready-to-latch limit switches indicate that the uplock rollers have engaged the latches. The signals enable the uplock latches to drive closed, compressing the aerothermal seals around each door.

Nominally, the umbilical doors are closed manually by the crew following ET separation. This is accomplished via switches on panel R2. During an RTLS or TAL abort, the doors close automatically. Automatic door closure reduces crew operations during time-critical abort situations. For the RTLS abort, the doors close following ET separation in MM601. For a TAL abort, the doors are closed in MM304. Note that the software which closes the umbilical doors during a TAL is open loop. No check is made to verify that ET separation has occurred prior to closing the umbilical doors. The automatic close capability for TAL cases is a new function added with Flight Software Release OI-20 (CR89354). The software for closing the umbilical doors during a TAL is contained in the vent door sequence.

2.5.3.4 Main Propulsion System Umbilical Lines

The orbiter and ET umbilical connections provide paths of transfer for LH_2 and LO_2 between the orbiter and ET. The umbilical lines include a 2-inch GO_2 and GH_2 pressurization line, a 4-inch LH_2 recirculation line, and a 17-inch LH_2 and LO_2 feedline. See figure 2.5-2. The 2-inch lines at each attach point are used to maintain positive pressure in the LO_2 and LH_2 tanks. This is achieved by routing gaseous propellants to the tanks from the SSME's. The 4-inch recirculation line on the LH_2 side is used to circulate LH_2 through the SSME's prelaunch. This provides thermal conditioning for the main engines before start. A 4-inch line is not installed on the LO_2 side because LO_2 is expelled overboard. The largest umbilical line is the 17-inch LO_2/LH_2 feedline. LH_2 flows to the main engines through the port 17-inch feedline and LO_2 flows through the starboard line. At ET separation, the umbilical lines are separated using special disconnect hardware.

2.5.3.4.1 17-Inch Feedline Disconnect.- The 17-inch LO_2/LH_2 feedline disconnect is shown in figure 2.5-3. Each feedline disconnect consists of a pair of two-position, pneumatically actuated, flapper valves (one on each side of the orbiter/ET interface). A feedline disconnect latch locks the flappers in the open position during ascent, allowing propellant to flow from the ET to the orbiter. Post-MECO, the latch is unlocked to ready the flapper valves for closure. The feedline disconnect latches were not part of the original orbiter design but were added beginning with STS-26. They prevent possible valve closure during main engine operation. If a valve failed closed during this time, it would starve the main engines of propellant and result in catastrophic shutdown.

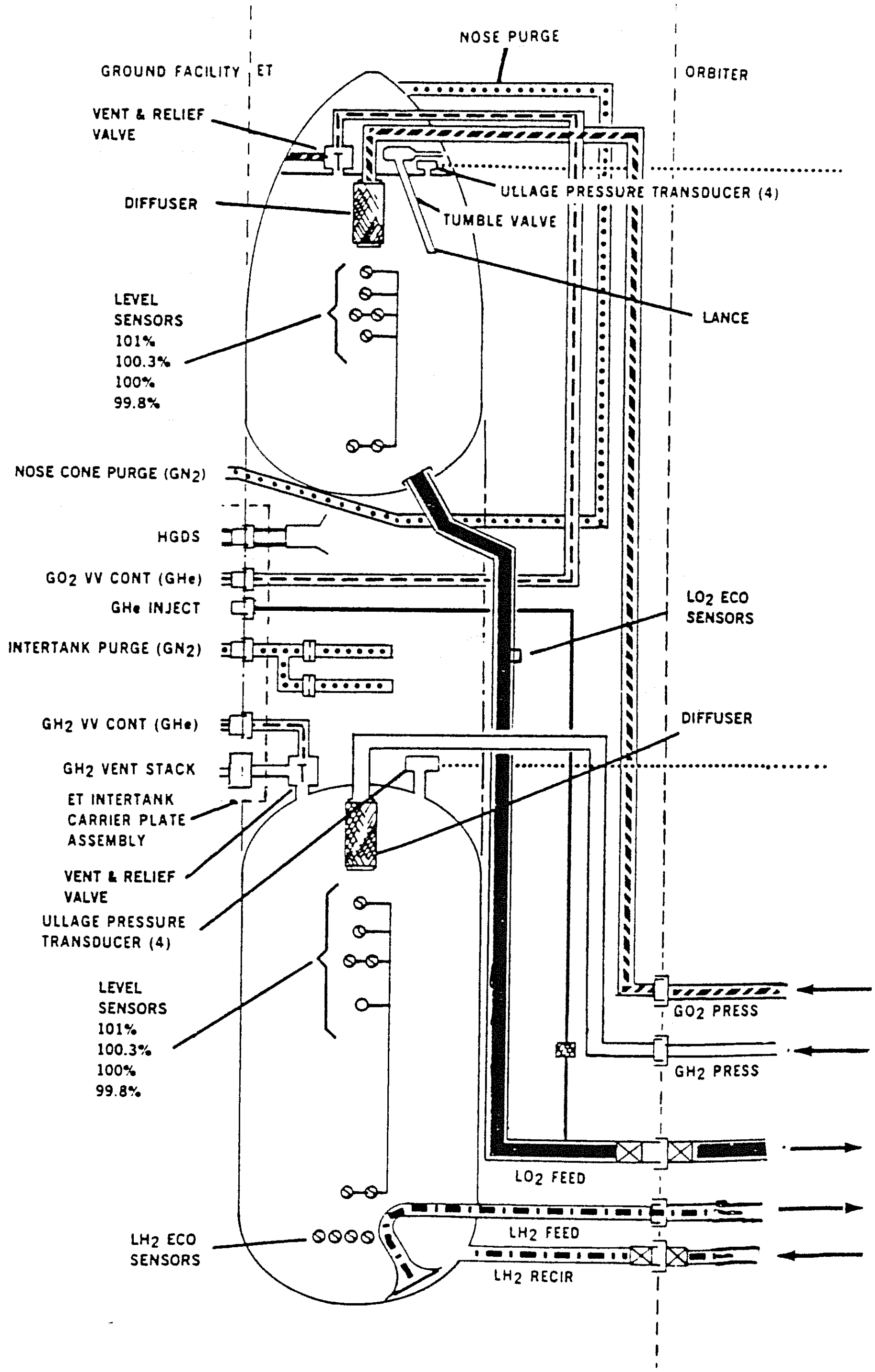


Figure 2.5-2.- External tank interconnect lines.

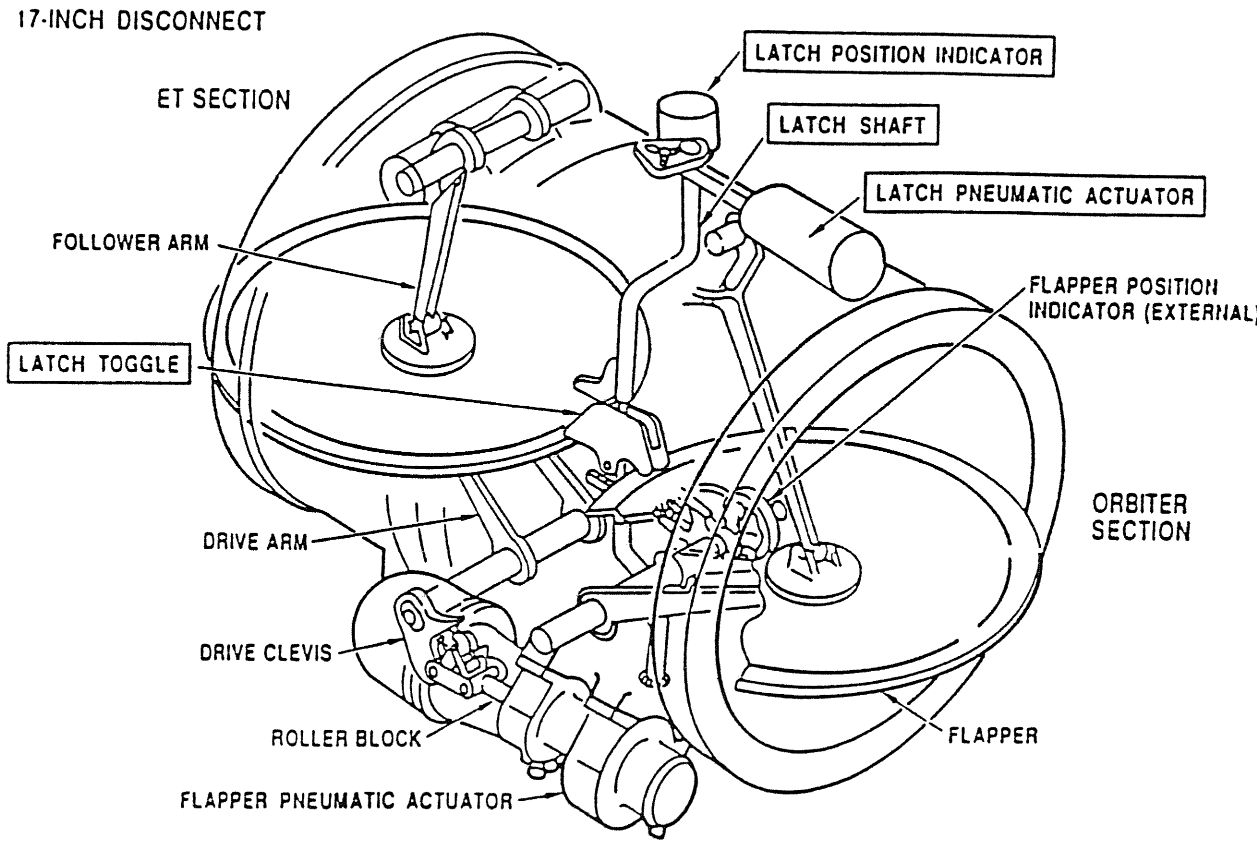


Figure 2.5-3.- 17-inch LO_2/LH_2 feedline disconnect.

There are two flapper valve position indicators for each 17-inch disconnect, feedback A and B. These redundant feedbacks are used by software to determine the position of the valves. The LH_2 line feedbacks are carried through FA2 and FA4 while the LO_2 feedbacks travel through FA2 and FA3. The MDM's that carry these feedbacks are significant because commfaults affect the ET separation process.

A secondary means of closing the 17-inch feedline disconnect flapper valves is provided through mechanical linkages. When the umbilical plate is retracted into the orbiter by the hydraulic retractors, a mechanical linkage system forces the valves closed. The mechanical system is especially important because it is the only means of closing the valves during "fast sep" ET separations.

The 17-inch disconnect splits in half during the separation process. One half of the line departs with the ET and the other half remains with the orbiter. The centerline of the disconnect is the separation plane, as shown in figure 2.5-4.

2.5.3.5 MPS Valves

Refer to figure 2.5-5 throughout this discussion.

2.5.3.5.1 Prevalves.- The MPS prevalves terminate flow of propellants to the main engines and help to safe the system. The SSME ops software commands the prevalves to close at MECO.

2.5.3.5.2 MPS LO_2/LH_2 Feedline Relief Isolation Valve.- The MPS LO_2/LH_2 feedline relief shutoff valve prevents propellant from escaping through the 17-inch feedline relief valve during SSME operation. The shutoff valve is opened post MECO and allows gasses trapped within the 17-inch line to reach the relief valve.

2.5.3.5.3 MPS LO_2/LH_2 Feedline Relief Valve.- The feedline relief valve allows propellants trapped in the 17-inch feedline to escape when pressure builds up to the relief setting. It prevents the line from overpressurizing and rupturing.

2.5.3.5.4 LH_2 RTLS Inboard and Outboard Dump Valves.- The LH_2 RTLS inboard and outboard dump valves provide the primary means of relieving pressure in the 17-inch feedlines post MECO. During a nominal ET separation, these valves are opened to provide pressure relief by exposing the feedline to the vacuum of space. In case of an RTLS abort, the 17-inch line is also purged with helium to assist in forcing LH_2 out the dump valves. This is necessary to ensure that the LH_2 is cleared from the system despite the higher ambient pressure experienced during an RTLS abort.

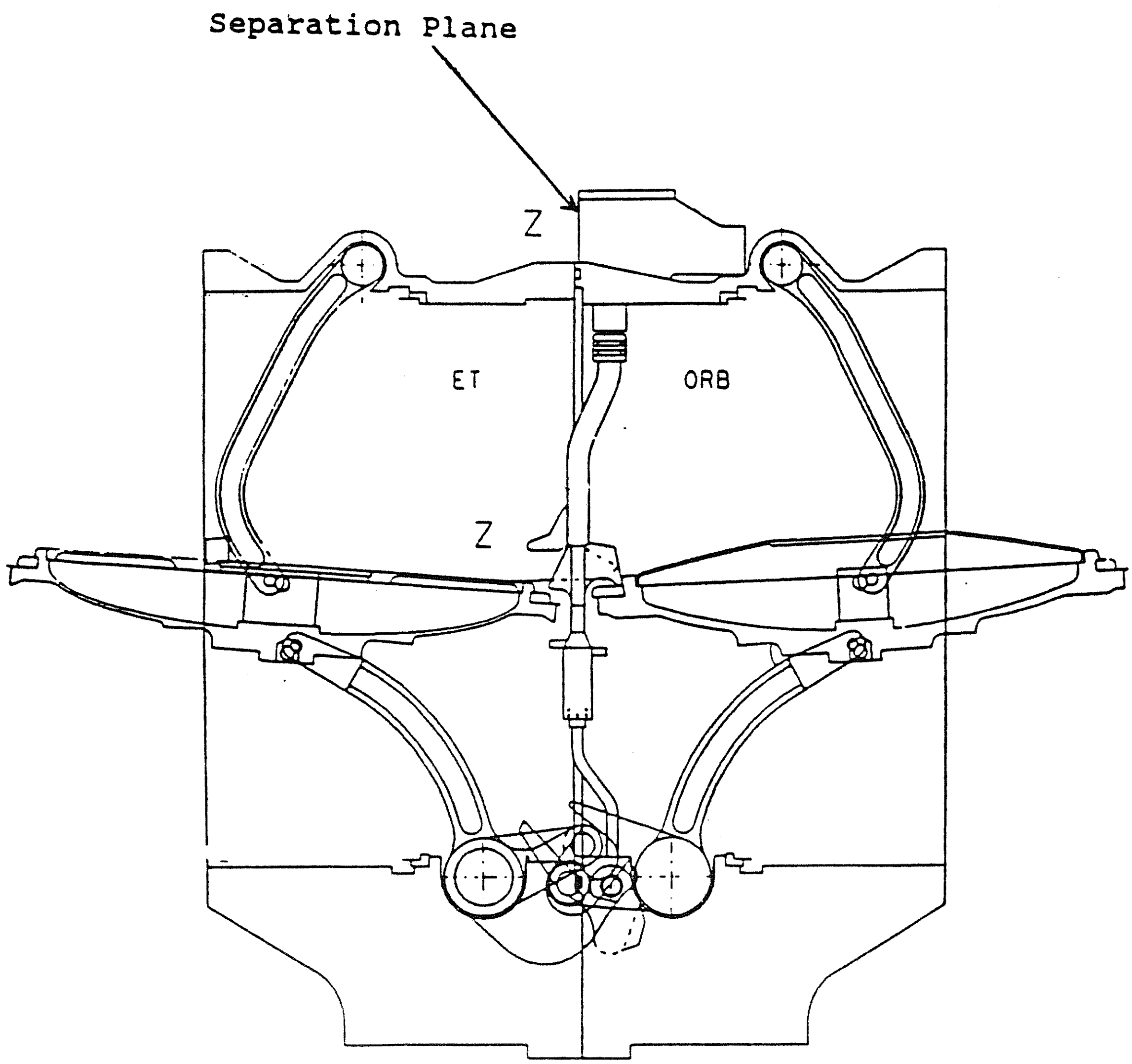


Figure 2.5-4.- 17-inch LO_2/LH_2 feedline disconnect.

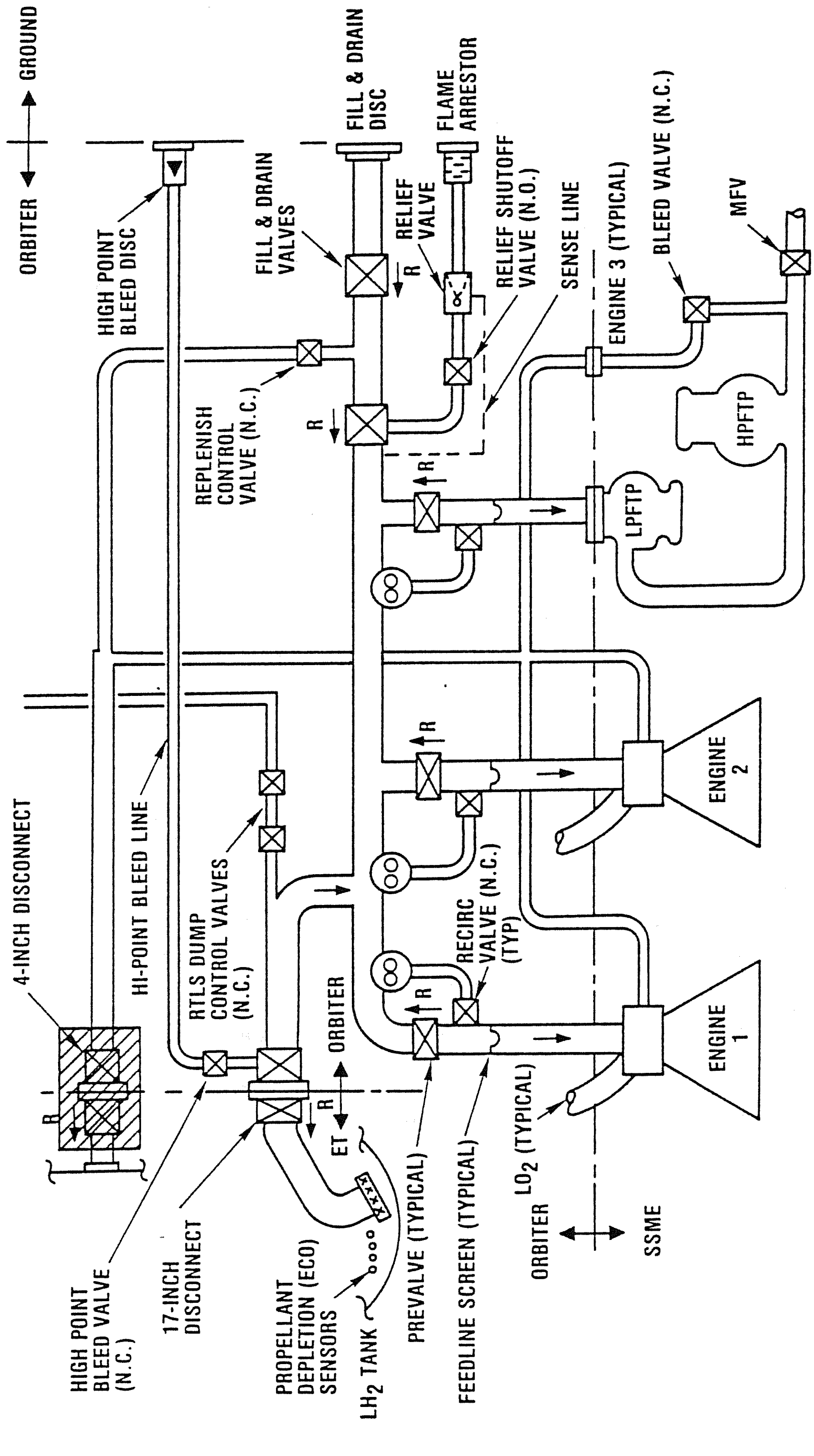


Figure 2.5-5.- MPS LH2 system.

2.5.3.6 ET Tumble Valve

The ET tumble valve was part of the original orbiter design but is no longer in use. It is located at the top of the ET and was designed to allow a small amount of GO_2 to escape from the tank following separation. See SSSH drawing 10.10. The escaping GO_2 caused the tank to begin a slow end-over-end tumble. The tumbling action was intended to ensure that the ET would follow a predetermined footprint during reentry. However, the tumble valve was found to be of minimal value in maintaining the ET footprint and had several risks associated with it. If the valve failed open during ascent, it could result in a catastrophic explosion. Therefore, the tumble valve hardware is disabled prior to each flight. External tanks currently in production do not include a tumble valve. However, ET separation software still contains the tumble valve commands.

2.5.4 ET Separation Software

ET separation is a complex process achieved through the interaction of numerous software modules. These include the ET separation sequencer, the transition digital autopilot (trans DAP), the GN&C moding, sequencing, and control (MSC) sequencer, the master events controller (MEC) SOP, and the SSME OPS module. Most of the separation activities are accomplished by just two of these modules, the ET separation sequencer and the trans DAP. An overview of the software interfaces during ET separation follows.

The MSC module schedules execution of the ET separation sequencer at the correct time in the ascent profile. Nominally this occurs when MECO is confirmed. The SSME OPS module sets the MECO confirmed flag and other discretes that are required by the MSC to achieve proper scheduling. The SSME OPS also provides flags to the ET separation sequencer, which indicate the mode of separation that is required during contingency abort cases. Physical separation of the orbiter and ET is accomplished by the ET separation sequencer; it is responsible for preparing the orbiter for separation and then executing the physical separation. Three modes of separation can be accommodated: automatic (nominal or RTLS), manual, and fast ET separation. Each of these is discussed in detail later. Once separation from the ET has been achieved, the trans DAP performs an automatic -Z translation away from the tank. It also monitors vehicle rates and determines when the translation should be terminated. The automatic separation process can be manually overridden by the crew at any time. Figure 2.5-6 illustrates the interactions between the modules.

Each software module involved in the separation process is discussed in detail in the following sections.

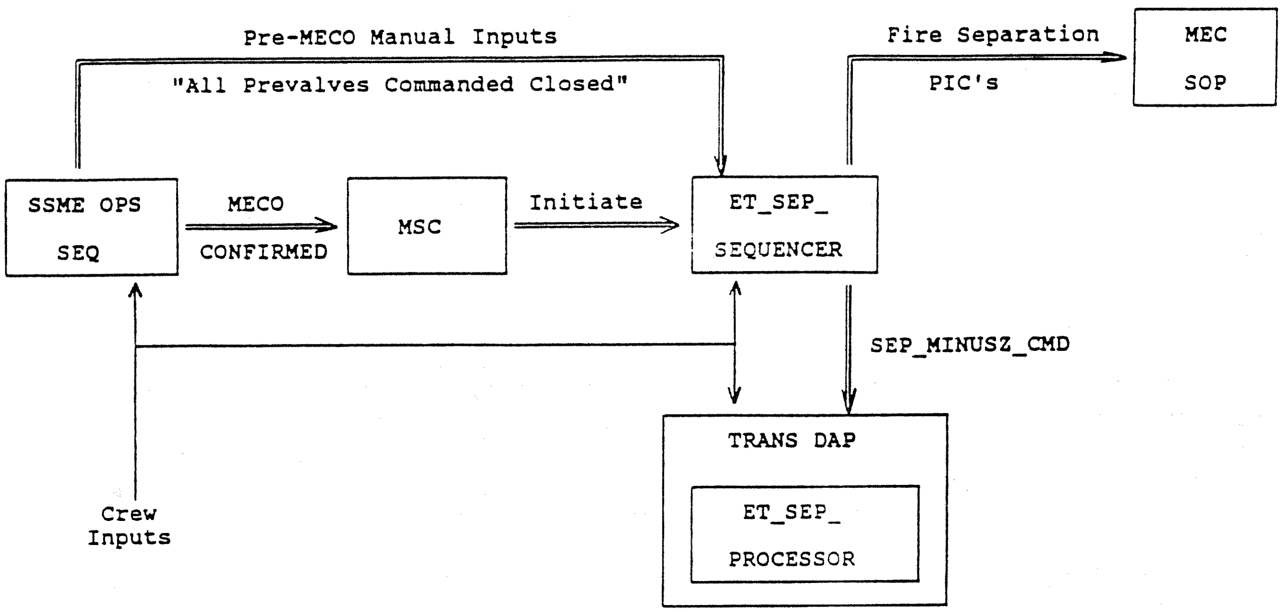


Figure 2.5-6.- ET separation software interfaces.

2.5.4.1 SSME OPS Sequencer

The SSME OPS sequencer serves several important functions in the ET separation process. First, it sets the MECO confirmed flag. This flag is set when indications from main engine sensors confirm that the engines have shut down. The MECO confirmed flag represents the beginning of the ET separation processes and is used by both the MSC sequencer and ascent DAP software. It indicates to the MSC sequencer that it is time to schedule execution of the ET sep sequence module. The ascent DAP uses this flag as an indication to terminate operation and hand orbiter control over to the trans DAP.

Another function of the SSME OPS module is to monitor manual ET separation discretes during MM102, MM601, and pre-MECO in MM103. Manual inputs during these periods affect the mode of separation executed by the ET sep sequencer. However, the sequencer is not normally active during these early phases of flight and cannot monitor for manual discretes. A fast separation is performed if the crew initiates manual ET separation in MM102 or MM601. It is also performed in MM103 if the crew initiates separation after two main engines have failed. The SSME OPS sends the appropriate indications to the ET sep sequencer if a fast separation is requested. It also sets the appropriate flags and delays to provide the proper SSME shutdown sequence for these cases. A fast sep can be performed in the BFS in OPS 102 only.

The final function of the SSME OPS sequencer, as far as ET separation is concerned, is to initiate timers for closing the LO_2 and LH_2 prevalves following MECO. The timers allow the engine to be shutdown safely by ensuring that the prevalves are closed at the proper time. When these timers expire, the SSME OPS sequencer commands the prevalves closed and sets the "All Prevalves Commanded Closed" discrete. This discrete is required by the ET sep sequencer to perform automatic ET separation correctly.

2.5.4.2 ET Separation sequencer

The ET sep sequencer is responsible for achieving physical separation of the orbiter and ET. It performs this function for nominal separations as well as for all abort cases. The activities it performs during nominal, ATO, AOA, and TAL aborts are essentially the same. RTLS abort activities differ significantly. This module may be active during ascent first stage (MM102), ascent second stage (MM103), OMS 1 insertion (MM104), OMS 2 insertion (MM105), insertion coast (MM106), or during an RTLS (MM601). The sequencer then operates cyclically at 6.25 Hz until just after structural separation in nominal, AOA, ATO, and TAL missions or until the umbilical doors are closed and latched in an RTLS.

Numerous functions must be performed to prepare the orbiter for ET structural separation. All of these functions are accomplished by the ET sep sequencer. It determines the mode of separation or, if the separation is to be manually inhibited, it closes the feedline disconnect valves, gimbals the SSME nozzles to the proper position, deadfaces the ET/orbiter interface, and unlatches and retracts the umbilical plates. The sequencer then arms the structural separation PIC's, performs limit tests on roll, pitch, and yaw body rates, angle of attack, slideslip angle, and tests for feedline disconnect valve closure before continuing with an automatic separation. If no inhibit conditions exist, the sequencer fires the structural separation PIC's and issues commands to the trans DAP to begin the -Z translation away from the tank.

The following discussion provides a step-by-step description of the sequencer as it performs each of these functions. It focuses primarily on activities during a nominal ET/orbiter physical separation. Some discussion is also provided regarding sequencer activity during manual, fast, and RTLS separations. It may be helpful to refer to information provided previously in the hardware section to understand the activities of the software.

2.5.4.2.1 Nominal (TAL, ATO, AOA) Automatic ET/Orbiter Physical Separation.- Physical separation of the orbiter and ET normally occurs automatically in MM103 and requires no special initialization from the crew. The sequencer proceeds through each of the steps described below to accomplish ET separation. The entire process normally requires approximately 18 seconds. Table 2.5-I provides a summary timeline of the ET separation process.

It should be noted that the numbers assigned to each of the following steps do not correspond to the step numbers given in the FSSR or DDS. The intention of this discussion is to describe events in the order in which they occur during a nominal separation. The actual software accomplishes this task through fairly complex control logic. Numbers assigned to the steps in the software do not necessary indicate the order in which the steps are executed. Therefore, the steps have been renumbered here to eliminate confusion.

Step 1 - Open Feedline Relief Shutoff Valve

The MPS LH_2/LO_2 feedline relief shutoff valves are commanded open. This allows propellants trapped in the 17-inch feedline to reach the feedline relief valves. These valves relieve pressure within the feedline when pressure builds up to the relief valve pressure limit. Pressure can build up in the line when the main engine prevalves and feedline disconnect valves are closed later in the sequence.

Step 2 - Unlock Umbilical Door Centerline Latches

The umbilical door centerline latches are unlocked. This allows for subsequent closure of the umbilical closeout doors.

TABLE 2.5-I.- ET SEPARATION SUMMARY TIMELINE

Time (sec)	Sequence of events for nominal separation	Software module
0	MECO command	
0.960	MECO confirmed	SSME ops
1.079	ET separation sequence initiated	MSC
	Feedline relief shutoff valve opened	Sep seq
	Unlock umbilical door centerline latches	Sep seq
4.710	Terminate ascent DAP, initiate trans DAP	Trans DAP
4.196	"All Prevalves Commanded Closed" discrete set true	SSME ops
	Unlock 17-inch feedline disconnect latches	
	Arm ET tumble system	
7.317	Allow prevalves to close	Sep seq
	Separation camera ON	Sep seq
	Tumble valve enabled	Sep seq
	Monitor 17-inch feedline latch positions	Sep seq
	Close 17-inch feedline disconnect valves	Sep seq
	Position main engines	Sep seq
9.717	Arm ET umbilical unlatch PIC's	Sep seq
11.317	Allow ET umbilical unlatch PIC's to charge	Sep seq
	Deadface electrical connections	Sep seq
	Fire ET umbilical unlatch PIC's	Sep seq
	Fire ET umbilical retract PIC's	Sep seq
	Open LH_2 RTLS inboard and outboard drain valves	Sep seq
16.917	Allow umbilical plate to retract	Sep seq
	Arm the structural separation PIC's	Sep seq
18.517	Check automatic separation inhibits	Sep seq
	Check for manual overrides	Sep seq
	Issue "SEP MINUS Z" command to trans DAP	Sep seq
18.677	Orbiter/ET physical separation	Sep seq
	Thrusters ON (four forward and six aft)	Trans DAP
	Issue "ET SEP CMD" flag	Sep seq
	MPS red and amber status lights turned off	DDS
	Zero MEC's output buffers	Sep seq
25.000	"SEP_COMPLETE" flag	Trans DAP
	MM104 transition	Trans DAP
91.63	Close LH_2 RTLS inboard and outboard dump valves	Sep seq
	ET separation sequencer terminated	

Step 3 - Monitor for "All Prevalves Commanded Closed"

The ET sep sequencer then waits for the "All Prevalves Commanded Closed" discrete to be set by the SSME OPS sequence. At MECO, the SSME OPS sequence initiates timers for closing the LO_2 and LH_2 prevalves. These times allow for a safe shutdown of the main engines by ensuring that the prevalves are closed at the proper time. When the timers have expired, the prevalves are commanded closed and SSME OPS sets the "All Prevalves Commanded Closed" discrete.

Step 4 - Unlock 17-inch Feedline Disconnect Latches

Once the ET sep sequencer receives the "All Prevalves Commanded Closed" discrete, it commands the 17-inch LO_2 and LH_2 feedline disconnect latches to unlock. This frees the feedline disconnect flapper valves in preparation for their closing. The sequencer also sends a command to arm the ET tumble system.

Step 5 - Allow Prevalves to Close

The sequencer pauses 1 second after receiving the "All Prevalves Commanded Closed" discrete to allow time for the prevalves to close and the feedline disconnect latches to unlock. This ensures that the disconnect valves are closed under conditions of no propellant flow.

Step 6 - Enable Separation Camera On/Tumble Valve

The separation camera located on the orbiter umbilical plate is commanded ON (OV-102) and the ET tumble valve enabled command is set.

Step 7 - Monitor 17-inch Feedline Latch Positions

The sequence now monitors the 17-inch feedline disconnect latch indications to determine if the latches have closed. A voting scheme is used to determine the latch position.

Each feedline disconnect, LO_2 and LH_2, has two redundant indicators for the latch locked and unlocked positions; i.e., locked A and B and unlocked A and B. The sequencer looks at each discrete and its associated commfault. It determines if the locked discretes are OFF and the unlocked discretes are ON. If any of the indications are commfaulted, the sequencer assumes that the commfaulted system is indicating the locked position. When any three of the four discretes indicate that the latches are unlocked, the sequencer assumes that the feedline disconnect flapper valves are unlatched. The three out of four voting allows the ET sep sequence to proceed with one indication failed.

If failures or commfaults cause the sequencer to receive less than three unlocked indications, a manual separation is required. A secondary means of closing the flapper valves is provided during umbilical plate retract, as described in step 14.

Step 8 - Close 17-inch Feedline Disconnect Valves

Having confirmed that the 17-inch feedline disconnect valves are unlatched (LH_2 and LO_2), the sequencer commands these valves to close.

Step 9 - Position Main Engines

The sequencer sends a flag to the MPS TVC CMD SOP to position the SSME nozzles in the MPS dump positions. In the event of a TAL or RTLS abort the main engines are commanded to their entry stow positions. The K-loaded values for the MPS dump and the entry stow positions are currently the same.

Step 10 - Arm ET Umbilical Unlatch PIC's

The sequencer waits for 2.3 seconds to expire after the LO_2 and LH_2 feedline disconnect valves were commanded closed. It then sets the "ET Umbilical Unlatch PIC Arm" Flag. The PIC arm flag signals the MEC to arm the ET umbilical unlatch PIC's.

Step 11 - Allow ET Umbilical Unlatch PIC's to Charge

The sequencer waits for 1.5 seconds to elapse after it issued the "ET Umbilical Unlatch PIC Arm" command. This pause allows the PIC's to become fully charged. The pause also allows a total of 3.8 seconds to elapse after the feedline disconnect valves were commanded closed (3.8 sec = 2.3 sec (step 10) + 1.5 sec (step 11)). The 3.8-second time period is a spec requirement to allow time for the feedline disconnect valves to become fully closed.

Step 12 - Deadface Electrical Connections

The sequencer deadfaces the electrical connectors between the orbiter and ET. This step involves resetting the ET developmental flight instrumentation (DFI) power and terminating the main propulsion system signal conditioner power.

Step 13 - Fire ET Umbilical Unlatch PIC's

The ET umbilical unlatch PIC's are fired. Detonation of these PIC's severs the three nut/bolt assemblies mating the orbiter and ET umbilical plates.

Step 14 - Fire ET Umbilical Retract PIC's

The ET umbilical retract PIC's are fired. Recall that these PIC's are not explosive devices. They power solenoid valves that allow hydraulic fluid to flow to the three hydraulic retractors on each umbilical plate. The hydraulic retractors draw the umbilical plates 2.5 inches into the orbiter. The retraction disconnects the orbiter/ET electrical umbilical in the first one-half inch of travel. The retract motion also closes the 17-inch feedline disconnect valves if they failed to close when commanded in step 8.

Finally, it releases fluids trapped between the orbiter and the ET 17-inch disconnect valves. Only two hydraulic actuators are required to retract the umbilical plates successfully.

Step 15 - Open LH_2 RTLS Inboard and Outboard Dump Valves

The LH_2 RTLS inboard and outboard dump valves are opened to relieve the pressure buildup caused by trapped propellants in the feedline. The dump valves are the primary means of relieving pressure in this line. In step 1 above, the MPS LH_2/LO_2 feedline relief shutoff valve was opened to provide a backup system to the dump valves. If the dump valves fail to open, pressure in the LH_2 manifold builds to the relief valve setting and the relief valve opens. This prevents the LH_2 feedline from rupturing.

Step 16 - Allow Umbilical Plate to Retract

The sequencer then waits 5.5 seconds from when it commanded the ET umbilical retract PIC's to fire (step 14). This delay allows time for the LO_2 and LH_2 umbilical plates to fully retract.

Step 17 - Arm the Structural Separation PIC's

The sequencer issues a command to the master events controller (MEC) to arm the ET/orbiter structural separation PIC's. The sequencer pauses 1.5 seconds to allow time for the PIC's to become fully charged.

Step 18 - Check Automatic Separation Inhibits

The sequencer checks for automatic separation inhibits, which includes testing the position of the LO_2/LH_2 17-inch feedline disconnect valves. These valves were commanded closed in step 8 (they should have also been closed mechanically when the umbilical plates retracted in step 14). Each flapper valve is equipped with two position feedbacks. At least one feedback must indicate the closed position, and not be commfaulted, for each flapper valve in order to proceed with automatic separation. The sequencer also checks that roll, pitch, and yaw body rates are within ±0.7°/sec. These rates were established to prevent recontact between the ET and orbiter. If any of the above tests are not satisfied, the separation is inhibited. An "ET SEP INH" message and a class 3 alert light and tone are annunciated to notify the crew. Separation cameras are also turned off to conserve film. The sequencer does not proceed until the out-of-tolerance parameter comes back within tolerance or the crew elects to continue the separation by manually overriding the inhibit.

Step 19 - Check for Manual Overrides

If all automatic separation criteria are satisfied, the sequencer makes a final check of the ET SEP AUTO/MAN switch position. If the switch is in AUTO, it proceeds with the automatic separation. All inhibit and manual override checks are bypassed following this step.

Step 20 - Issue "SEP MINUS Z" Command to Trans DAP

The Sequencer issues the "SEP MINUS Z" command to the trans DAP. This discrete commands the trans DAP to begin the orbiter -Z translation which separates it from the ET. The "SEP MINUS Z" command is issued before actual orbiter/ET physical separation in order to allow 160 ms for the trans data to configure for the -Z RCS burn.

Step 21 - Orbiter/ET Physical Separation

The sequencer commands the separation camera on. This command is needed in case the camera was turned off earlier because of automatic separation inhibits. It then commands the orbiter/ET structural separation PIC's to fire. This command causes the MEC SOP to fire the separation PIC's, separating the orbiter from the ET. The sequencer also issues a discrete, the "ET SEP CMD" flag, to other software modules signifying that ET separation has occurred. This causes the MSP dedicated drive sequencer to turn off the MPS red status lights that were turned on at MECO. It also causes G/C steer to initiate steering commands for ET separation. In a TAL abort, it causes the vent door control software to command all vent doors to the closed position to prevent ingestion of propellant during the propellant dump. Finally, a timer is started that prevents the ET umbilical door centerline latches from stowing for 2 seconds after physical separation. This delay prevents any movement of the umbilical doors or latches until the ET is clear of the orbiter.

Step 22 - Zero MEC's Output Buffers

The sequencer commands the MEC SOP to zero the MEC's critical command output buffers and reset all of the noncritical fire 3 commands.

Step 23 - ET Umbilical Door Closure

Umbilical door closure normally is accomplished manually by the crew via switches on panel R2 (excluding RTLS and TAL aborts). The doors are closed manually to allow the crew to monitor the door closure process and take appropriate action if a problem is observed. When the crew manually closes the doors, the software described in this step is bypassed.

In case of an AOA, time does not permit the crew to close the doors manually. The ET sep sequencer provides the option to close the umbilical doors automatically. The crew commands automatic door closure in OPS 1 by executing an item 20 on the GNC 51 override spec (PASS). Refer to Flight Rule 10-130. When the ET sep sequencer receives this command, it executes the following steps.

First, the ET door centerline latches are commanded to stow, assuming the 2-second timer that was started at structural separation has expired. The sequence allows 6 seconds for the latches to release and then commands the left and right ET doors to close. The sequencer waits another 48 seconds to allow time for complete door closure (single-motor time). The left and right uplock latches are then commanded to latch. Twelve seconds are allowed for these latches to close fully (single-motor time).

There is some risk involved in selecting the automatic umbilical door closure sequence described above. The sequencer issues commands based on nominal motor drive time and does not check or stop for anomalous operation. The potential exists to drive the doors against the centerline latches (if they do not stow properly), damaging the drive mechanism and preventing proper ET door closure. This risk is considered acceptable during an AOA.

In the case of an RTLS or TAL abort, umbilical door closure is accomplished automatically without any commanding required by the crew (no spec item entries). The doors are closed by the ET sep sequencer during an RTLS abort using a sequence similar to the one described above. During a TAL abort, the umbilical doors are closed in MM304 by software contained in the vent door sequencer.

Step 24 - Close LH_2 RTLS Inboard and Outboard Dump Valves

The sequencer waits for the expiration of an 80-second timer before closing the LH_2 RTLS inboard and outboard dump valves. These valves were opened in step 15 to relieve feedline pressure. The original orbiter software set the value of this timer to only 23 seconds. It was updated with the OI8C flight software version as a result of CR 89399LB. The additional time increases system reliability against overpressurization. The CR also revised sequencer logic so that the MEC master reset would be done before expiration of the timer instead of after. This ensures that the MEC is reset before MM304 transition in case of a TAL abort.

Step 25 - ET Sep Sequencer Terminated

The ET sep sequencer has completed its functions and is terminated.

2.5.4.2.2 Manual Separation.- The crew has the capability to manually inhibit ET separation at any point or to manually initiate the separation in the presence of automatic inhibits. This capability is provided by the ET SEP AUTO/MAN switch and the SEP pushbutton or item entries on SPEC 51. This was described earlier in the crew interface section. In the following discussion, it is assumed that manual separation has been declared and one of the following conditions is true:

1. It is MM103 and "MECO Confirmed" has been set

2. It is MM104, 105, or 106

If manual separation is initiated under other than these conditions, a fast separation or RTLS separation is executed.

Before the initiation of manual separation, the ET sep sequencer proceeds with automatic separation as described above. Throughout the automatic separation process, the sequencer continuously monitors for manual inputs. When the ET SEP switch is placed in the MAN position, the sequencer commands the ET separation cameras OFF and waits for the SEP pushbutton. It does not proceed further in the sequence until the pushbutton is engaged. The cameras are commanded off to conserve film in the event that the automatic sequence had reached a point where the cameras were turned on.

When the SEP pushbutton is engaged, the sequencer resumes the ET separation process where it was interrupted when the ET SEP switch was taken to MAN. The sequence then proceeds as it would for an automatic separation except for one important difference. The check for automatic separation inhibits is bypassed (step 18 above). The sequence proceeds directly to step 19 and issues the "SEP MINUS Z" command to the trans DAP. All sequencer activities proceed as described above for the automatic separation.

2.5.4.3 Transition Digital Autopilot

The trans DAP is one of six digital autopilots that provide flight control throughout the numerous flight regimes of the shuttle. The primary responsibility of the trans DAP is to provide control logic during insertion and deorbit phase of flight. For a detailed description of the trans DAP see the GNC Transition Digital Autopilot Systems Brief.

The first duty of the trans DAP when it assumes orbiter control following MECO is to perform ET separation. The trans DAP is active during a NOM, TAL, AOA, and ATO ET separation. It serves three major functions in the separation process. First, it maintains orbiter rate and attitude deadbands throughout the various phases of the separation. Second, it executes the orbiter -Z translation, assuming no automatic inhibits or crew overrides. Third, it monitors vehicle rates during separation and issues the "SEP COMPLETE" flag, signifying completion of the separation process.

The separation process can be divided into two distinct phases, the mated coast phase and the -Z translation phase. The -Z translation represents the only automatic reaction control system (RCS) translation of the mission. Each phase of separation is described in detail below.

2.5.4.3.1 Mated Coast.- The trans DAP assumes orbiter control from the ascent DAP following issuance of the "MECO Confirmed" flag by the SSME OPS sequencer. When the ascent DAP receives the "MECO Confirmed" flag, it initiates an attitude hold. The trans DAP maintains this attitude hold when it assumes orbiter control 3.75 seconds later. This begins the mated coast phase. Attitude and rate deadbands during this period are ±10° and ±0.5°/sec, respectively.

While the trans DAP maintains the orbiter within the specified deadbands, the sep sequencer performs the functions described in the previous section. The sep sequencer nominally requires approximately 18 seconds to complete its functions, assuming no automatic inhibits or crew overrides. Throughout the mated coast period, the trans DAP monitors for the "SEP_MINUSZ_CMD" from the sep sequencer. This flag is the signal from the sequencer that it has completed preparations for ET/orbiter physical separation. Upon receiving this flag, the trans DAP initiates the -Z translation.

2.5.4.3.2 -Z Translation.- When the trans DAP receives the "SEP_MINUSZ_CMD" from the sep sequencer, two events occur. First, the trans DAP takes a snapshot of the orbiter attitude and maintains that attitude throughout the -Z translation. Second, commands are issued to the RCS jets to begin firing for the -Z translation. The jet commands come from a dedicated module within the trans DAP called the ET_SEP_PROCESSOR. This module commands firing of 10 -Z translational jets, four forward and six aft. The jets begin to fire at the same time the ET sep sequencer commands the ET structural separation PIC's to detonate. These PIC's shear the nut/bolt assemblies mating the orbiter and ET. A minimum number of thrusters is required in each thruster group to ensure that orbiter/ET recontact does not occur during the translation. Flight Rule 8-5 provides guidelines for the minimum number of RCS jets required.

At initiation of the -Z translation, attitude control is inhibited for 2.96 seconds. This is done to prevent attitude control jet activity from reducing orbiter translation rates during the critical period immediately following physical separation. The attitude hold is implemented by temporarily severing communication between attitude control modules in the trans DAP and the RCS jets. This allows attitude control modules to continue running during the 2.96-second inhibit and prevents the need for system reinitialization at resumption of attitude control. Attitude control is resumed by reenabling communication between control modules and thrusters.

At the end of the attitude control inhibit period, the trans DAP maintains attitude deadbands of ±3.0° and rate deadbands of ±0.5°/sec. Throughout the translation, the DAP does not fire any jets in opposition to the -Z jets. Trans DAP jet select logic prevents upfiring jets from being commanded while a -Z translation is commanded. To correct errors in pitch, downfiring -Z translation jets are turned off.

Typically, the trans DAP turns off two forward downfiring jets immediately following the 2.96-second attitude control inhibit to keep the orbiter within its pitch rate deadband. Analysis shows that firing the 10 -Z translation jets for 2.96 seconds causes a positive pitch rate of approximately 0.5 deg/sec. The rate results from the unbalanced moment arms of the forward and aft jets.

As the -Z translation proceeds, the ET_SEP_PROCESSOR monitors orbiter translation rates. It compares the orbiter translation rate it receives from the ascent/entry attitude processor to an I-loaded value. When the -Z translation rate reaches this value, the ET_SEP_PROCESSOR issues the "SEP_COMPLETE" flag. The current value of this I-load is 4 ft/sec for a NOM, AOA, or ATO separation and 11 ft/sec for a TAL abort separation.

The "SEP_COMPLETE" flag signifies termination of the ET_SEP_PROCESSOR and notifies other trans DAP software modules that separation is complete. The trans DAP then automatically modes to MM104 and configures itself for the coming OMS burns. Automatic transition to MM104 is the crew's indication that the separation process is complete. Post-ET separation attitude deadbands are ±3.5° and rate deadbands are ±0.5°/sec in all axes.

2.5.4.3.3 Manual -Z translation.- The crew normally allows the -Z translation to proceed automatically. However, they have the ability to assume manual control of the translation at any time. This may become necessary in certain failure scenarios or to execute a detailed test objective (DTO). Manual control is activated by moving the translational hand controller (THC) out of detent. It is not reversible. Taking manual control causes the ET_SEP_PROCESSOR to terminate its -Z translation jet commands. However, it continues to monitor vehicle rates. It is the crew's responsibility to see that the I-loaded -Z translation rate is achieved. When the orbiter reaches this rate, the ET_SEP_PROCESSOR issues the "SEP_COMPLETE" flag and the trans DAP automatically modes to MM104.

Taking the rotational hand controller (RHC) out of detent or moding the DAP to MAN via the DAP PBI panel, before or during the -Z translation, does not prevent the -Z burn from occurring automatically.

If the crew does not achieve the I-loaded -Z translation rate, the ET_PROCESSOR does not issue the "SEP-COMPLETE" flag. There are two consequences of this. First, the trans DAP does not automatically transition to MM104. The crew has to manually execute the transition. Second, the impact is seen in the deadband maintained by the trans DAP. Attitude deadbands of 3° remain in effect instead of the 3.5°g deadbands normally in effect following ET separation. In such cases, the "SEP_COMPLETE" flag is issued when an OMS engine is commanded to fire or OPS 1 is exited.

2.5.4.3.4 Fast Sep.- A fast sep is a contingency separation procedure that greatly reduces the time between ET SEP CMD and structural separation. A fast sep is performed if the ET SEP PB is depressed and either 1) MM102 or 601 or 2) MM103 and ≥2 SSME's failed in turn.

2.5.5 References

Space Shuttle External Tank Systems Definition Handbook Configuration and Operation, Volume 1
MMC-ET-SEP25-0
DR SE25 WBS 1.6.2.2
NAS8-30300
Revision B
August 1987

Booster Systems Briefs
Systems Division, Guidance and Propulsion Systems Branch
Basic, Rev B
April 1, 1990

Sequencing Requirements, Rational and Historical Reports
Rockwell International
Space Transportation Systems Division
STS90D0307

Functional Subsystem Software Requirements (FSSR), Level C
Guidance, Navigation, and Control, OI-20
STS83-0026C (Sequence Requirements)
September 28, 1990

Functional Subsystem Software Requirements (FSSR), Level C
Guidance, Navigation, and Control, OI-20
STS80-0008C, Volume 2 (Trans DAP)
June 18, 1990

Space Shuttle Programs Orbiter Avionics Software Operational (OBS) Detailed Design Specification (DDS)
OI20 Publication
Volume III - Applications
Part 1: Guidance, Navigation & Control (GN&C)
3.8.3.5 External Tank Separation Sequence (GST_ET_SEP)

Master Events Controllers 2102 Workbook
Johnson Space Center Training Library
February 9, 1979

Mechanical Systems 2102 Workbook
National Aeronautics and Space Administration
November 1989

3.1 MPS OVERVIEW

3.1.1 Function of the Main Propulsion System

The MPS consists of the plumbing network located in the rear of the orbiter, aft of the payload bay. This plumbing network furnishes the SSME's with liquid propellants from the ET. The MPS overview is shown in figure 3.1-1.

The SSME's, assisted by two SRB's during the initial phases of the ascent trajectory, provide the vehicle velocity increment (delta V) from lift-off to a predetermined velocity before orbital insertion. Once the SRB's burn out at approximately 2 minutes and are jettisoned (first stage), the MPS continues to draw propellants (LO_2 and LH_2) from the ET so that the SSME's may continue to burn during second stage. At predetermined velocity, the SSME's are shut down (MECO) and the ET is jettisoned. Then, for standard insertion missions, the liquid hydrogen manifold is vented for 6 seconds and the OMS is ignited to provide the final delta V required for orbital insertion (OMS 1). The magnitude of the delta V supplied by the SSME's will depend on the payload weight, mission trajectory requirements, and system limitations.

During standard insertion missions, the remaining MPS propellant will be dumped coincident with the start of the OMS 1 burn. During direct insertion missions (no OMS 1 required), the MPS propellant dump will be performed 2 minutes after MECO. This prevents the MPS propellant from contaminating the area of the orbiter that may house an experiment or package. This also ensures that the MPS plumbing is not damaged by the trapped propellants. Later, after the propellant dump, the pilot will vacuum inert the MPS to ensure that the MPS manifolds will not build up pressure while the orbiter is on orbit.

During entry, the MPS propellant lines are repressurized with helium to prevent contaminants from being drawn into the lines. At this time, helium is also used to purge the aft compartment. The last activity involving the MPS will occur at the end of the landing rollout, at which time the onboard helium storage tanks will be blown down to safe the system.

3.1.2 MPS Subsystems

The following paragraphs give a brief description of the MPS subsystems. The subsystem interfaces are shown in figure 3.1-2.

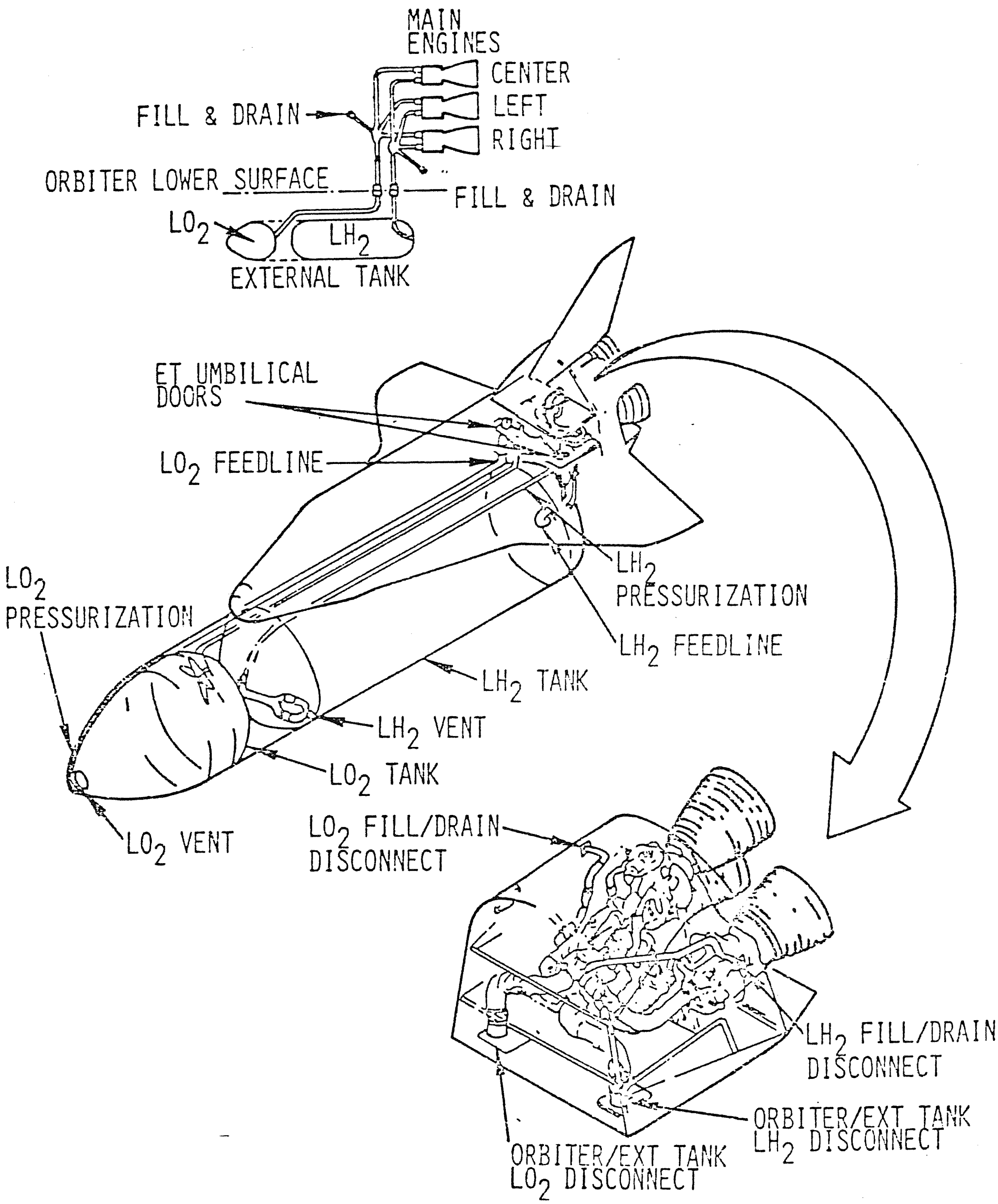


Figure 3.1-1.- Main propulsion system.

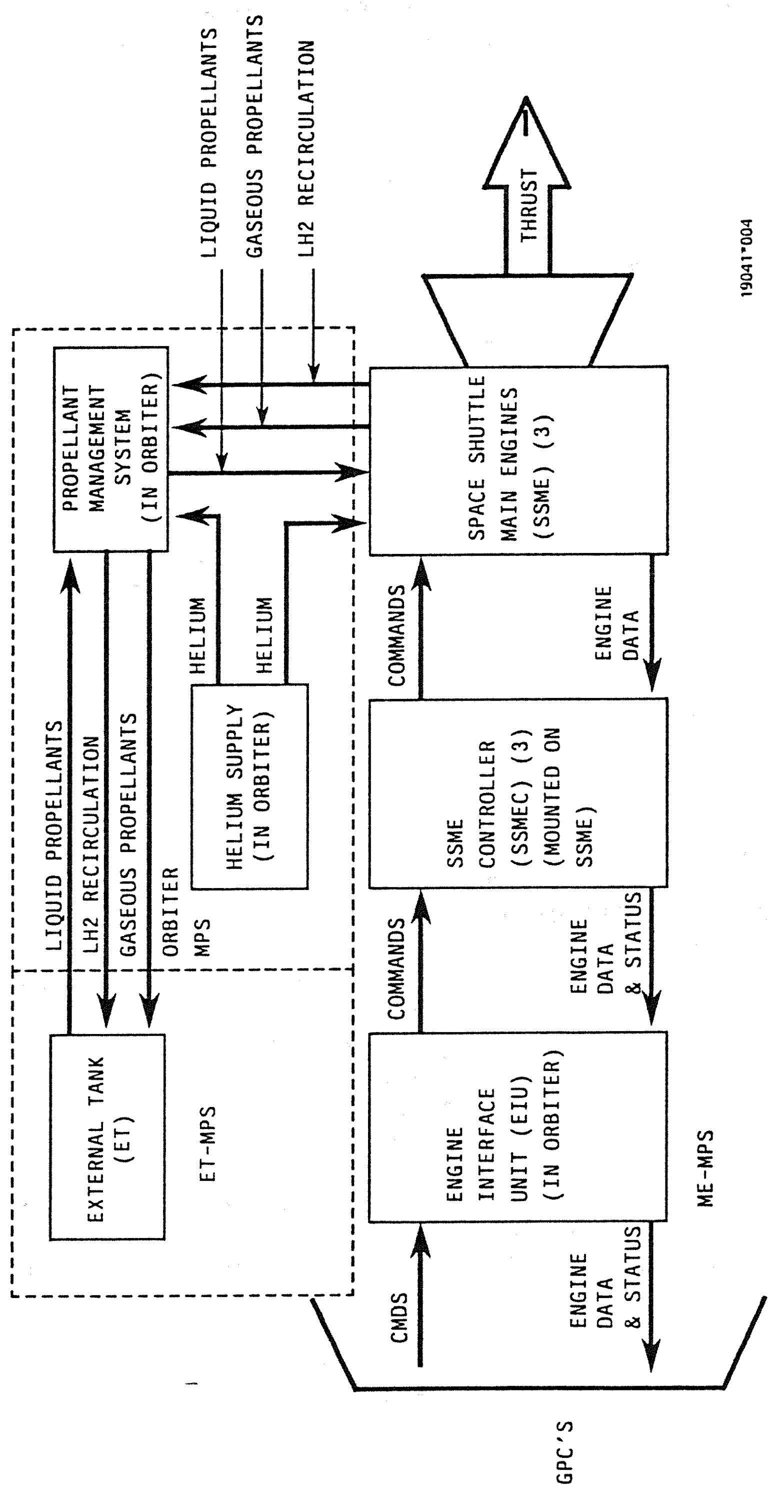


Figure 3.1-2- The MPS subsystem interface.

3.1.2.1 MPS Helium Supply

The helium supply furnishes helium for critical SSME operation, shutdown purges, the pneumatically actuated MPS valves, and for the pressurized dump post-MECO.

3.1.2.2 Orbiter MPS-Propellant Management System

All of the orbiter MPS is located in the compartment aft of the payload bay. The propellant management system furnishes the SSME's with liquid propellants (LO_2 and LH_2) from the ET. The MPS also routes gaseous oxygen and hydrogen from the SSME's to the ET for tank pressurization.

A schematic of MPS interfaces with other vehicle systems is shown in figure 3.1-3.

3.1.3 MPS Propellant Flow

An overview of the MPS propellant flow is shown in figure 3.1-4.

The LO_2 and LH_2 propellant manifolds consist of the 17-inch lines that receive propellants from the ET. Each line then gets split into three 12-inch lines (one per engine). The manifold consists of the 17-inch pipe and the reduction into three 12-inch lines to feed the engines. Each manifold has a pressure sensor that the crew can see on the panel F7 meter and BFS SYS SUMM 1 display. Both manifolds have propellant fill and drain valves that allow filling the ET during prelaunch. These are also used in flight for dumps and vacuum inerting. There are two valves in series and an overboard vent. The crew has switches on panel R4 that control these valves. Each manifold has a relief valve leading to an overboard vent. Each manifold relief valve can be isolated by its feedline relief isolation valve. The crew can control the isolation valves from panel R4. Each manifold ends where it meets the LO_2 and LH_2 pre-valves for each engine. The manifolds each receive helium from the helium system to repressurize the manifold or to expel propellant during the dump sequence. The LH_2 manifold differs from the LO_2 manifold because it has two valves in series that allow venting the LH_2 manifold overboard. These are the backup LH_2 dump valves (also known as the RTLS LH_2 dump valves). The crew can open and close them from panel R2 only when the GPC's are in major mode 1. The topping valves manifold is used during propellant loading to ensure the proper engine temperatures and a full ET tank. The topping valve manifold connects the LH_2 bleed valve in each engine to the fill and drain line between the inboard and outboard fill and drain line valves. These manifolds and the topping valves are used during the normal dump sequence.

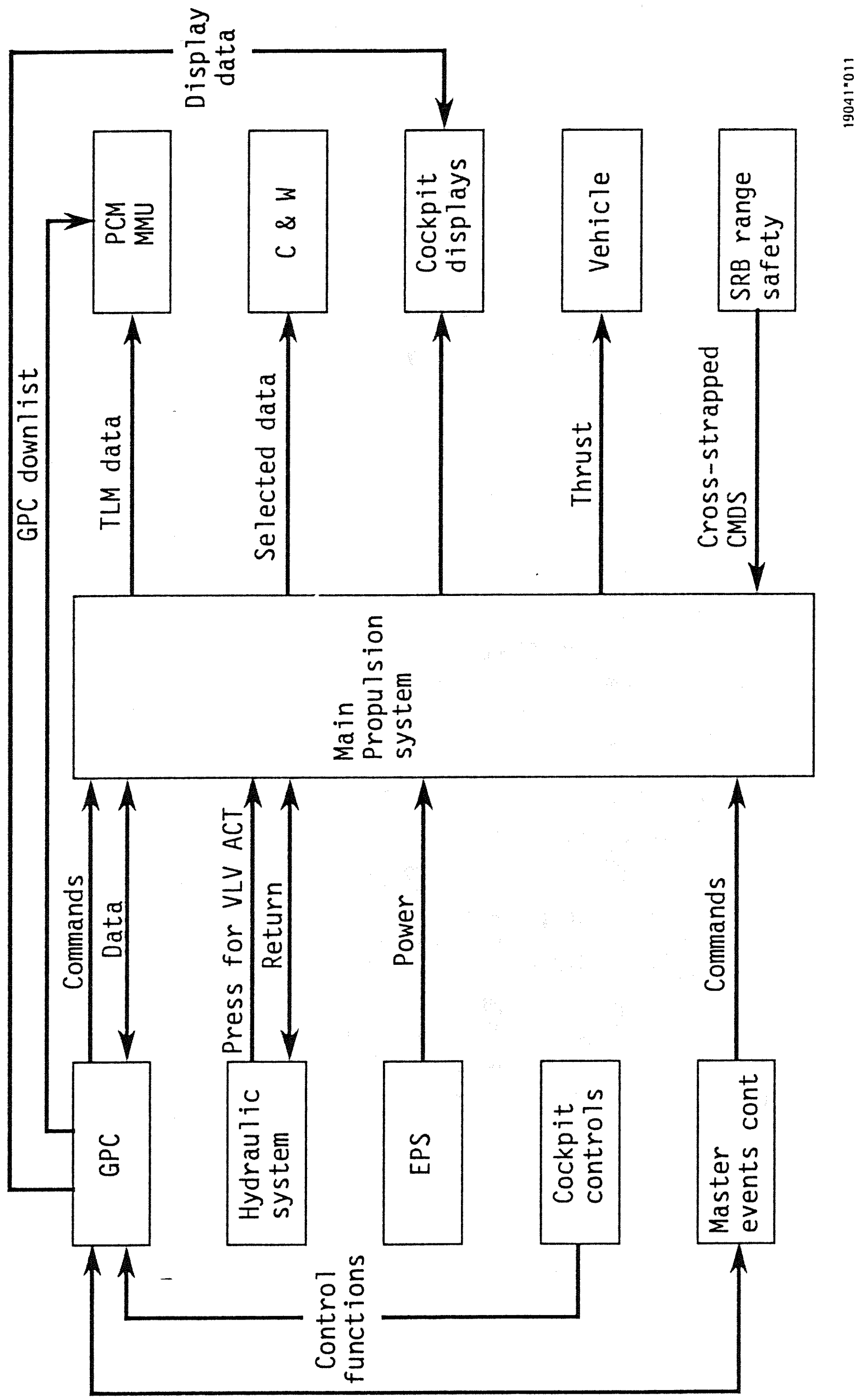


Figure 3.1-3- The MPS interfaces with other vehicle systems.

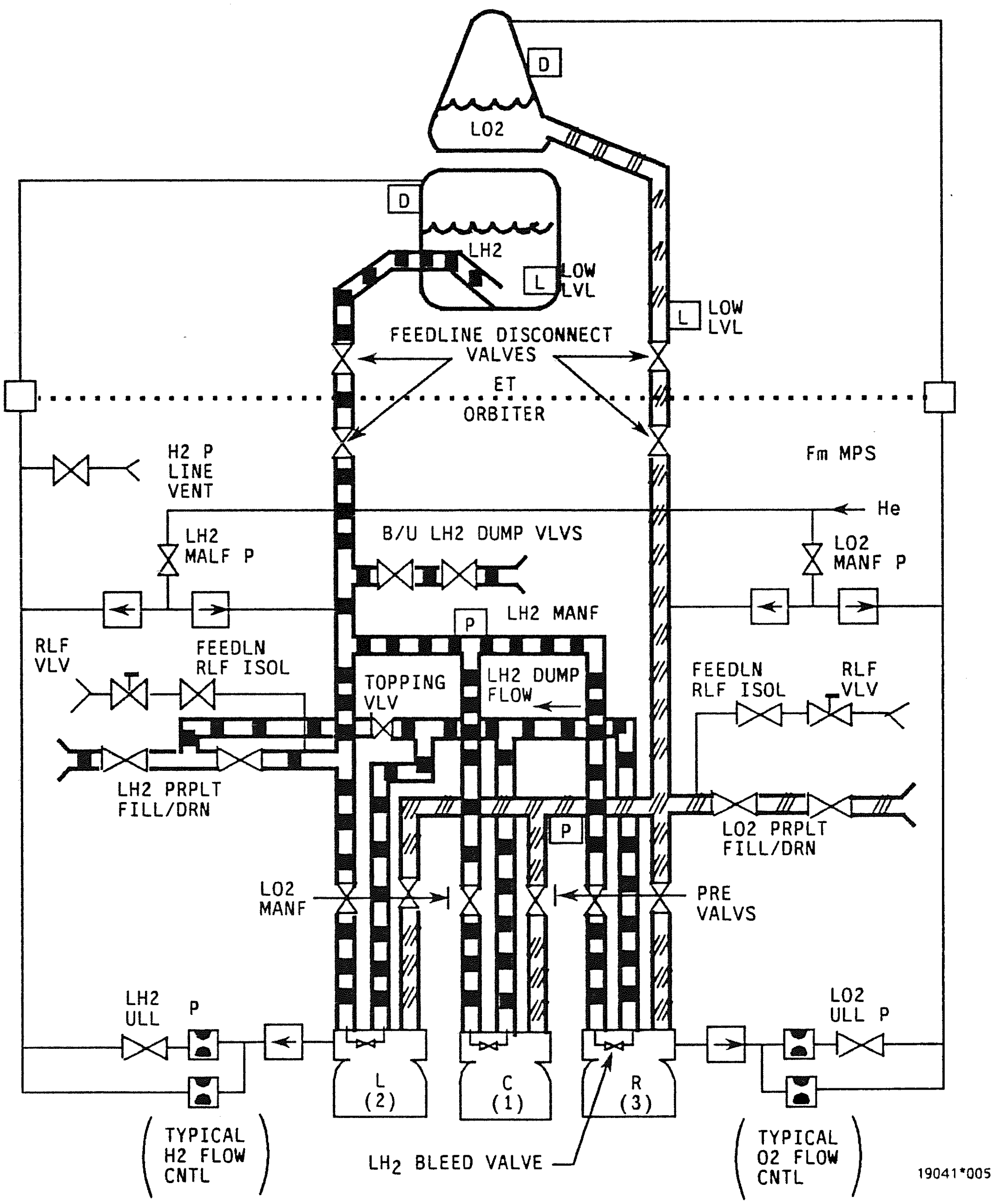


Figure 3.1-4.- MPS propellant flow.

3.2 MPS HELIUM SYSTEM DESCRIPTION

3.2.1 Introduction

This systems brief addresses the helium system and its uses. Helium is used in the MPS during flight for purge of the intermediate seal in the high pressure oxidizer turbopump, propellant valve pneumatic actuation, in-flight main engine purges, propellant line repressurization during entry and landing, and aft compartment blowdown.

3.2.2 General

The MPS helium system consists of seven 4.75 ft^3 tanks and three 17.33 ft^3 tanks. The tanks are connected into a pattern of three engine paths (left, center, and right) and one pneumatic path. Each engine path consists of one 17.33 ft^3 tank in the midbody area and two 4.75 ft^3 tanks, one in the mid-body and one in the aft fuselage. The pneumatic path consists of one 4.75 ft^3 tank in the aft fuselage.

The left, center, and right helium supply tanks are used to support the respective SSME during engine start, mainstage, and shutdown. The pneumatic helium supply is used to operate the main engine valves and MPS valves and to supplement engine purges during the shutdown sequence. If a main engine helium supply path leak is detected, the pneumatic helium supply can be manually interconnected to allow the engine to run longer. During entry, the remaining helium is used to repressurize the LO_2 and LH_2 manifolds (repress) and to purge the aft compartment of possible combustible gas concentrations such as H_2 and N_2H_4 (hydrazine). After rollout, helium is used to control SSME valve position when hydraulics are brought up for SSME repositioning (rain drain).

The 4.75 ft^3 and 17.33 ft^3 tank configuration is used in lieu of one single large tank for several reasons. First, demand for helium has evolved and grown as the space shuttle design was finalized. The 4.75 ft^3 tank in the aft compartment is an example of adding a tank to meet an additional SSME demand for helium. Second, space constraints are conducive to smaller multiple tanks rather than a single large tank. Finally, the smaller tanks were initially identified for space shuttle use and were readily available when additional tanks were needed.

3.2.3 Functional Description

Each of the three-engine systems consists of the following hardware below the helium supply tanks (fig. 3.2-1). Immediately downstream of the aft fuselage 4.75 ft^3 tank is the pressure transducer from which the cockpit CRT and helium meters are driven. This same pressure reading is brought into the control center and displayed on MPS MSK 1054 display (HE TK P). Below this pressure transducer, the system splits into two redundant legs (A and B). Each leg can be independently opened or closed with cockpit switches.

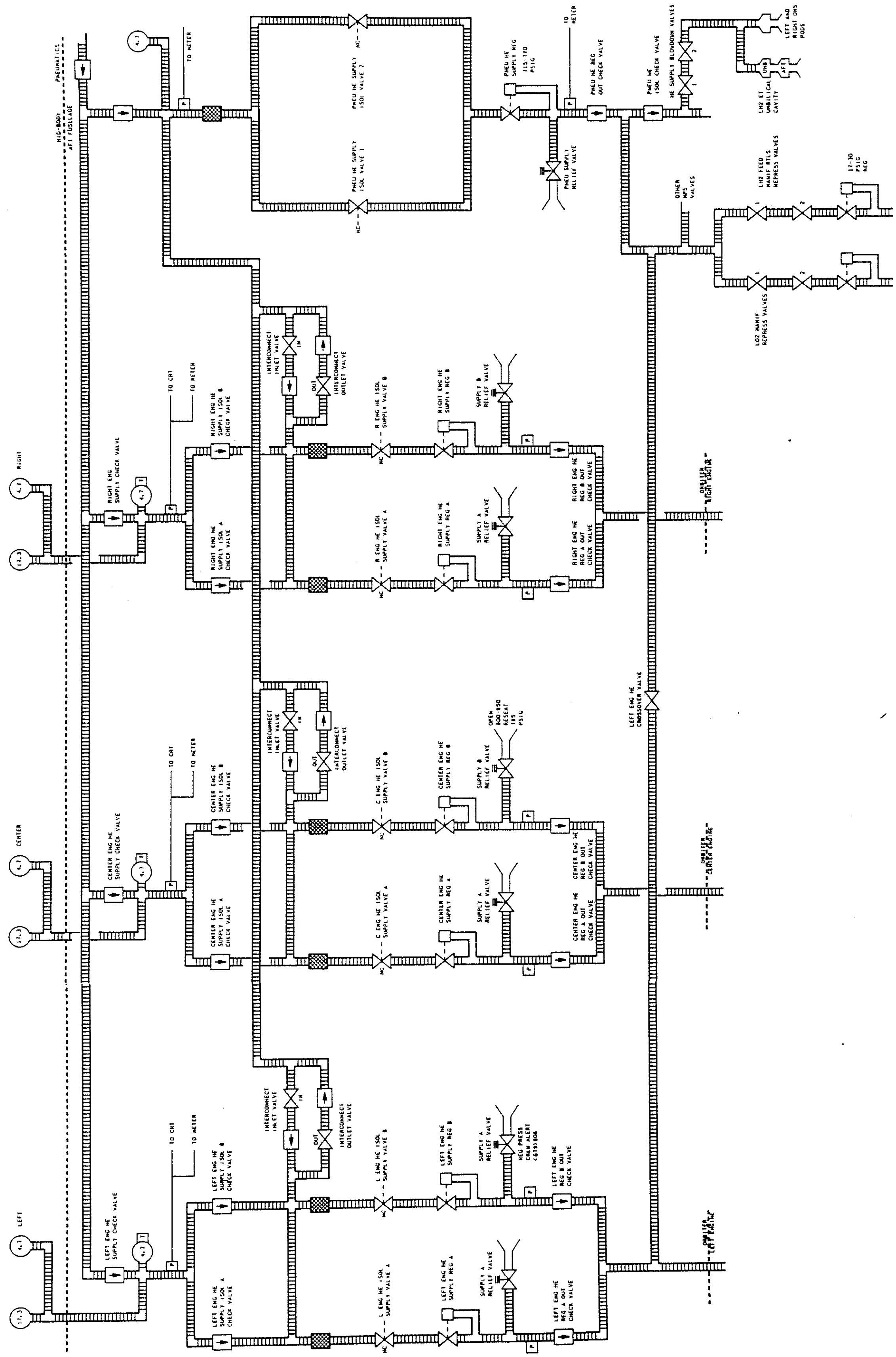


Figure 3.2-1.- MPS helium supply system.

Each leg contains a supply check valve, interconnect junction, filter, supply isolation valve, supply regulator, relief valve, regulator pressure transducer (HE REG P), and outlet check valve. The two legs are then reconnected into a single path leading to a main engine. The left engine system outlet has one unique valve, a crossover valve, that connects it to the pneumatics helium outlet to provide redundancy for the pneumatic helium supply. When this valve is opened, helium from the left engine system is allowed to flow into the pneumatics system below its regulator check valve. The interconnect valves allow helium to flow into or out of each engine system through manually opened inlet or outlet valves. These valves are normally closed on ascent, but can be opened to feed pneumatic helium to a running engine or to feed a running engine from a failed engine. The pneumatics system and the engine systems are essentially the same. Just downstream of the supply tank is the interconnect path.

The pneumatics helium is always available to the engine systems and the engine system inlet interconnect valves need only to be opened for helium to flow from the pneumatics system into an engine system. The pneumatics helium path uses a single filter before the path is split into two redundant legs (A and B), each containing an isolation valve. The pneumatic helium isolation valve configuration differs in two ways from the engine systems: the isolation valves are the only hardware on the redundant legs and the two valves are controlled by a single switch. After the two legs are merged again, the following hardware is encountered: supply regulator, relief valve, regulator pressure transducer, and outlet check valve. Below the outlet check valve is the junction of the left engine helium crossover line. Downstream of this junction is the pneumatic system isolation check valve and all of the MPS pneumatically operated valves. Specific details on any of the helium valves can be found in SB 3.3.

The MPS helium valves are controlled by cockpit switches on panels R2 and R4. The isolation, crossover, and manifold repress valve switches have three positions: open, close, and GPC. The interconnect valve switch positions are, in open/out close, out open/in close, and GPC. In the unpowered GPC position, both the inlet and outlet interconnect valves are closed. The switch position/valve status profile for the major MPS activities is contained in table 3.2-I.

TABLE 3.2-I.- (SWITCH POSITION) VALVE STATUS

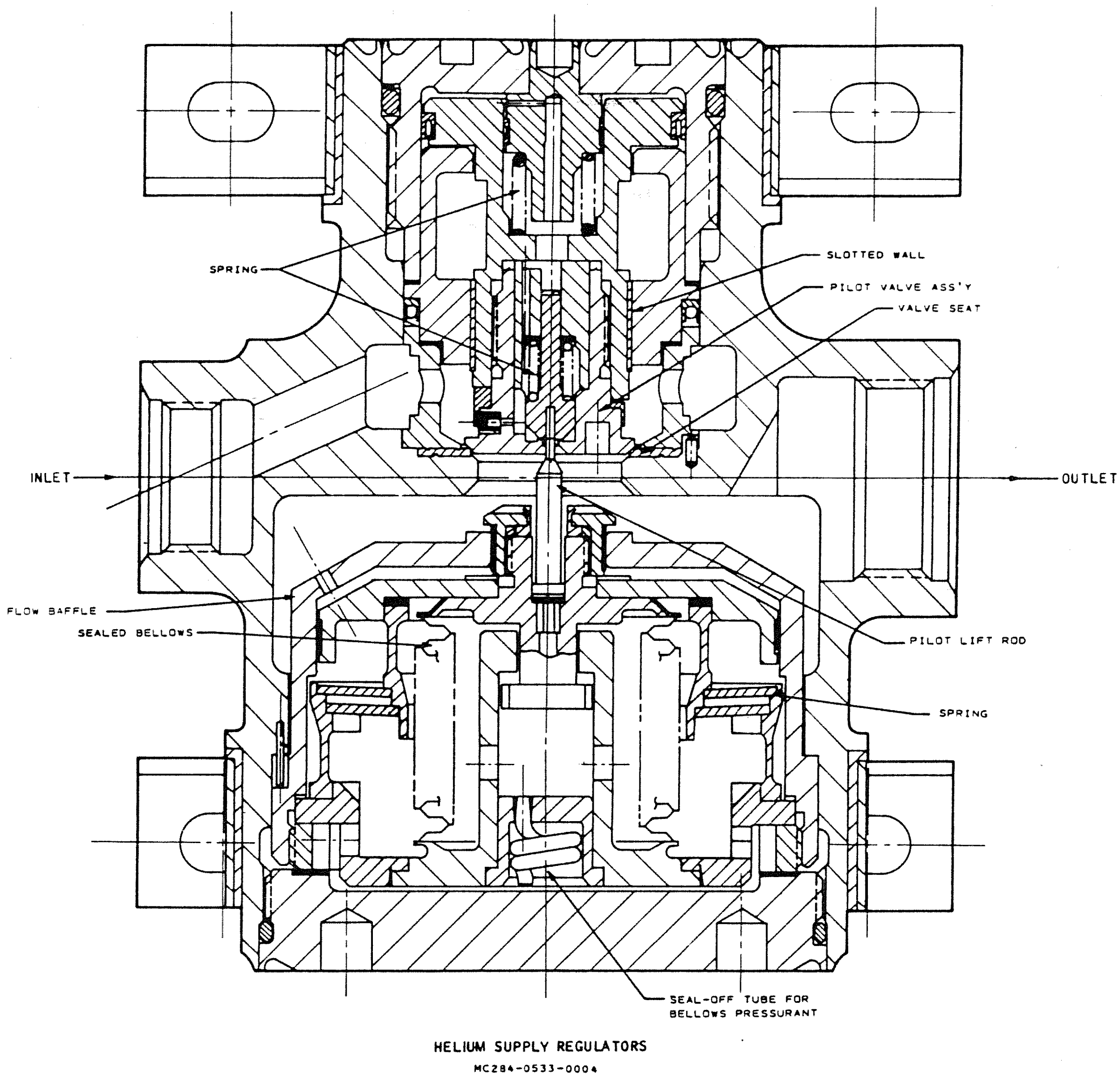
Activity / Valve	Prelaunch	Powered flight	ET sep	MPS dump	Vacuum inert	On orbit	Entry* purge MM304
Isolation valve L	(GPC) Open	(Open)** Open	(Open) Open	(Open) Open	(GPC) Closed	(GPC) Closed	A Open B Closed
C	(GPC) Open	(Open)** Open	(Open) Open	(Open) Open	(GPC) Closed	(GPC) Closed	A Open B Closed
R	(GPC) Open	(Open)** Open	(Open) Open	(Open) Open	(GPC) Closed	(GPC) Closed	A Open B Closed
P	(GPC) Open	(Open)** Open	(Open) Open	(Open) Open	(Open) Open	(GPC) Closed	(Closed) Closed
Interconnect valve L	(GPC) Closed	(GPC) Closed	(GPC) In open	(GPC) Out open	(GPC) Closed	(GPC) Closed	(GPC) In open
C	(GPC) Closed	(GPC) Closed	(GPC) In open	(GPC) Out open	(GPC) Closed	(GPC) Closed	(GPC) Out open
R	(GPC) Closed	(GPC) Closed	(GPC) In open	(GPC) Out open	(GPC) Closed	(GPC) Closed	(GPC) Out open
Left eng crossover	(GPC) Closed	(GPC) Closed	(GPC) Open	(GPC) Open	(GPC) Closed	(GPC) Closed	(GPC) Open
Manif repress (LO_2 & LH_2)	(GPC) closed	(GPC) Closed	(GPC) Closed	(GPC) Open	(GPC) Closed	(GPC) Closed	(GPC) Open
Blowdn valves***	Closed	Closed	Closed	Closed	Closed	Closed	Open

* All isolation valves set to manual (open/closed) and interconnect valves to GPC at TIG-25 min. Interconnect valves commanded open by GPC's at EI-5 min. Manifold repress and blowdown valves open at V_{REL}-4.5k. GPC's close blowdown valves post touchdown.

** Crew sets these positions at T-16 minutes.

*** No switch control.

A schematic of the helium supply regulators (i.e., the six SSME helium regulators and the pneumatic helium supply regulator) appears in figure 3.2-2. These operate at 715/770 psia and lock up at 785 psia, minimum. When the regulator outlet pressure drops below the regulator setting, the outlet compartment bellows will expand and the pilot lift rod will push open the main poppet. When the outlet pressure rises above the regulator setting, the bellows is compressed, the lift rod is retracted, and the main poppet is spring-loaded closed.



DASH NO.	USE
0004	1. E1,E2,E3 HELIUM SUPPLY REGULATOR 2. PNEU VALVE HELIUM SUPPLY REGULATOR

Figure 3.2-2.- Helium supply regulators.

Although the SSME A and B regulators are identical, they are used for different limit sensing purposes. Knowledge of these applications is essential to properly assess both nominal and off-nominal helium consumption. On the cockpit helium meter when it is configured to read regulator pressures, only the REG A values are displayed. Also, only REG A values are used in the caution and warning (C&W) system, and are limit sensed between 680 and 810 psia. In the systems management (SM) alert system, both REG A and REG B values are sensed between 679 and 806 psia. In all cases, a helium regulator high value warns of a failed open regulator. The low value warns of a relief valve failed open, a regulator shifted low, or a line break. A more thorough description of the C&W and SM alert systems can be found in section 2.3.1 of the Booster Console Handbook.

3.2.4 Predicted Consumption

The initial helium load is supplied from ground servicing equipment (GSE) during prelaunch with final topping occurring at T_0-13 seconds. After this point in the launch countdown, all MPS helium is onboard the orbiter. The minimum, nominal, and maximum helium system masses onboard at T_0-13 seconds is presented in table 3.2-II.

The following lists the nominal, maximum, and minimum expected helium system masses at T_0-13 seconds (GSE helium fill terminate).

TABLE 3.2-II.- MPS HELIUM MASSES

Case	Subsystem location	Pressure & temperature (psia)	(°R)	Volume (ft^3)	Mass (lb)
	SSME supplies				
	midbody	4270	582	66.24	165.41
Nominal	Aft-fuselage	4270	506	14.25	35.59
	SSME subtotal			80.49	201.00
	Pneu. supply	4320	525	4.75	12.63
	Total helium system			85.24	213.63
	SSME supplies				
	midbody	4500	505	66.24	193.00
Maximum	Aft-fuselage	4500	455	14.25	41.52
	SSME subtotal			80.49	234.52
	Pneu. supply	4500	455	4.75	14.81
	Total helium system			85.24	249.33
	SSME supplies				
	midbody	4100	595	66.24	156.51
Minimum	Aft-fuselage	4100	525	14.25	33.67
	SSME subtotal			80.49	190.18
	Pneu. supply	4100	590	4.75	12.05
	Total helium system			85.24	202.23

The expected consumption of helium for typical, abort-once-around (AOA), and return to launch site (RTLS) missions is shown in table 3.2-III. The typical and RTLS uses are shown for a nominal initial load mass.

TABLE 3.2-III.- HELIUM USAGE SCHEDULE (LB)

Mission	Typical	AOA	RTLS[1]
Helium load	NOM	NOM	NOM
Initial mass	213.63	213.63	213.63
Usage			
Prelaunch (T_0-13 → T_0)			
SSME purges	-3.02	-3.02	-3.02
Pneumatics	-0.56	-0.56	-0.56
Boost (T_0 → S/D+20)			
SSME purges	-70.09	-69.32	-125.00
Pneumatics boost leakage	-0.27	-0.27	-0.27
Pneumatics valve actuation	-0.50	-0.50	-0.50
Redundant engine shutdown	-0.04	-0.04	-0.04
Engine shutdown	-7.34	-7.34	-7.34
SSME baggy purge[2]	-0.46	-0.46	-0.46
Mass remaining at S/D +20	131.45	132.12	76.08
Post shutdown & orbital coast			
LO_2 propellant dump	-4.60	-4.60	-----
LH_2 propellant dump[3]	-6.07	-6.07	-6.07
Orbital coast leakage	-0.56	-0.07	-----
SSME baggy purge[2]	-0.01	-0.01	-0.01
Vacuum inert	-1.18	-1.18	-----
Reentry mass	118.93	120.19	70.00
Reentry purges[4]			
Umbilical CAV/AFT-fuselage	-39.05	-39.05	-39.05
OMS pods	-17.55	-17.55	-17.55
SSME baggy purge[2]	-0.52	-0.52	-0.52
Leakages (PNEU,SSME)	-0.02	-0.02	-0.02
MPS repress	-4.50	-4.50	-4.50
Residual helium (at rollout)[5]	57.29	58.55	72.80

[1]Includes loss of 1 SSME He supply at liftoff (66.0 lbm). RTLS burn time - 736 seconds. Nominal He usage - 233 SCFM.

[2]Assumes baggy purge of 2 SCFM.

[3]OI - 23 He usage; OI - 22 usage; Nom/AOA - 7.41; RTLS - 0.

[4]Purges for H2 17 Q/D UMB CAV/aft-fuselage and OMS pods purges are inadequate to keep hydrogen concentrations below 4% level during reentry.

[5]He is required for 4 hours after wheelstop for manifold repress.

The flow rate of helium during flight is not constant, but rather rises and falls in response to the various demands. Figure 3.2-3 depicts the nominal flow rate of helium in the three engine paths from T_0-16.6 seconds to SSME shutdown +18 seconds.

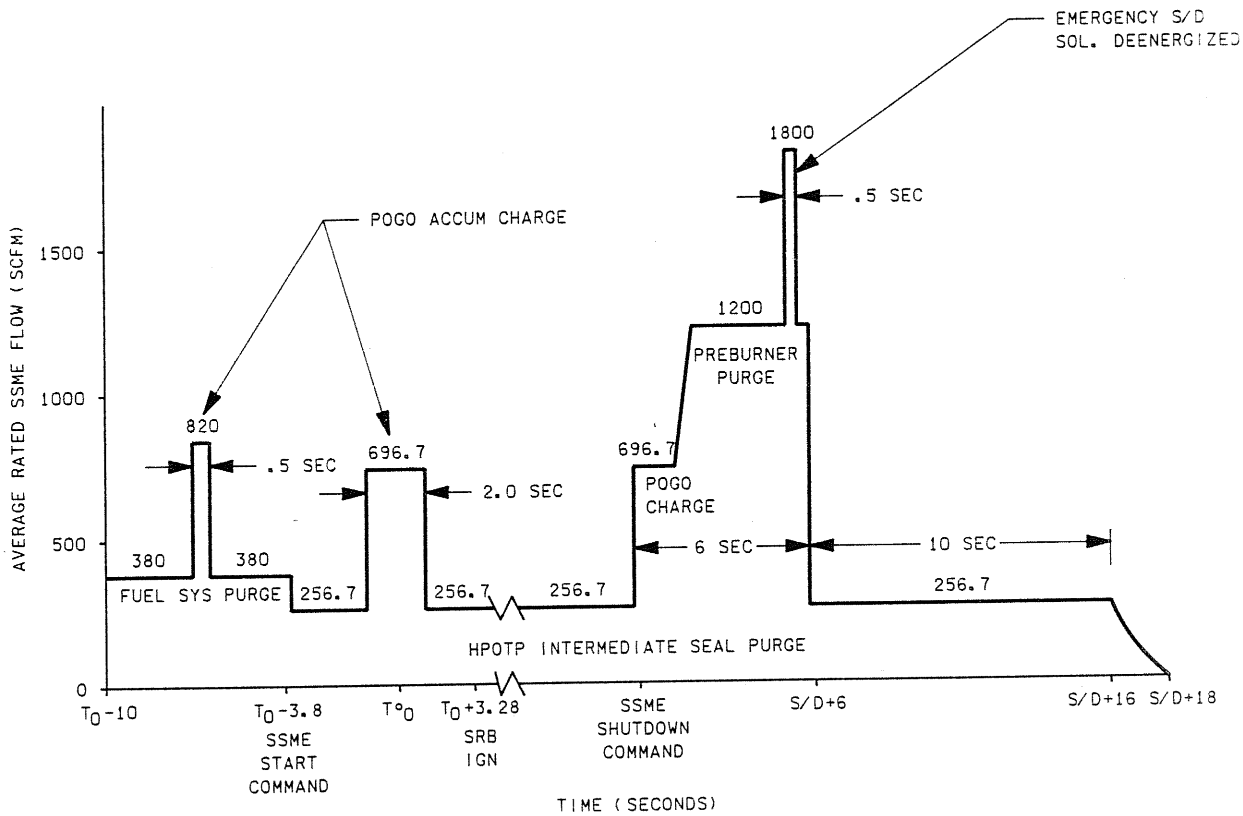


Figure 3.2-3.- SSME helium supply rated flow vs. time.

3.2.5 Flow Rates

Two units of measure are commonly used in discussions of helium flow, scfm and lb/sec. The following series of equations reveals the relationship between the two.

Flow (lb/min) = 0.0103 x flow (SCFM)
Flow (lb/sec) = 0.000172 x flow (SCFM)

For example, flow = 1000 SCFM
= 10.3 lb/min
= 0.172 lb/sec

A representative comparison of this relationship is shown in table 3.2-IV.

TABLE 3.2-IV.- SCFM-LBS/SEC CONVERSION*

SCFM	Lbs/sec
100	0.0172
200	0.0344
300	0.0516
400	0.0688
500	0.0860
600	0.1032
700	0.1204
800	0.1376
900	0.1548
1000	0.1720
1100	0.1892
1200	0.2064
1300	0.2236
1400	0.2408
1500	0.2580
1600	0.2752
1700	0.2924
1800	0.3096

*Standard conditions are 68° F and 1 atmosphere pressure.

3.2.6 Conclusion

This SB has described the basic arrangement and function of the MPS helium system and how the contents of that system are budgeted for use during flight. More specific information regarding malfunction procedures and computations can be found in the following SCP's in the Booster Console Handbook.

SCP No.	Title
2.2.4	Helium Leak Isolation/Interconnection
2.2.9	MPS Entry Helium Purging
3.2.2	SSME Helium Supply Computations
7.2.1	MPS On-Orbit Helium Remaining Computation

3.2.6 References

STS-6 Operational Orbiter MPS Helium System Performance and Margin assessment, Rockwell Internal Letter P&PP/MPS/82/009, 29 September 1982.

Shuttle Operational Data Book (SODB)

Booster Console Handbook, Final, Rev D, JSC-17239, Feb 29, 1988

3.3 MPS VALVES SCHEMATICS

3.3.1 General

The propellant management subsystem is a system of manifolds, distribution lines, and valves by which the liquid propellants pass from the ET to the SSME's and where gaseous propellants pass from the SSME's to the ET.

The helium subsystem has regulators, check valves, and control valves.

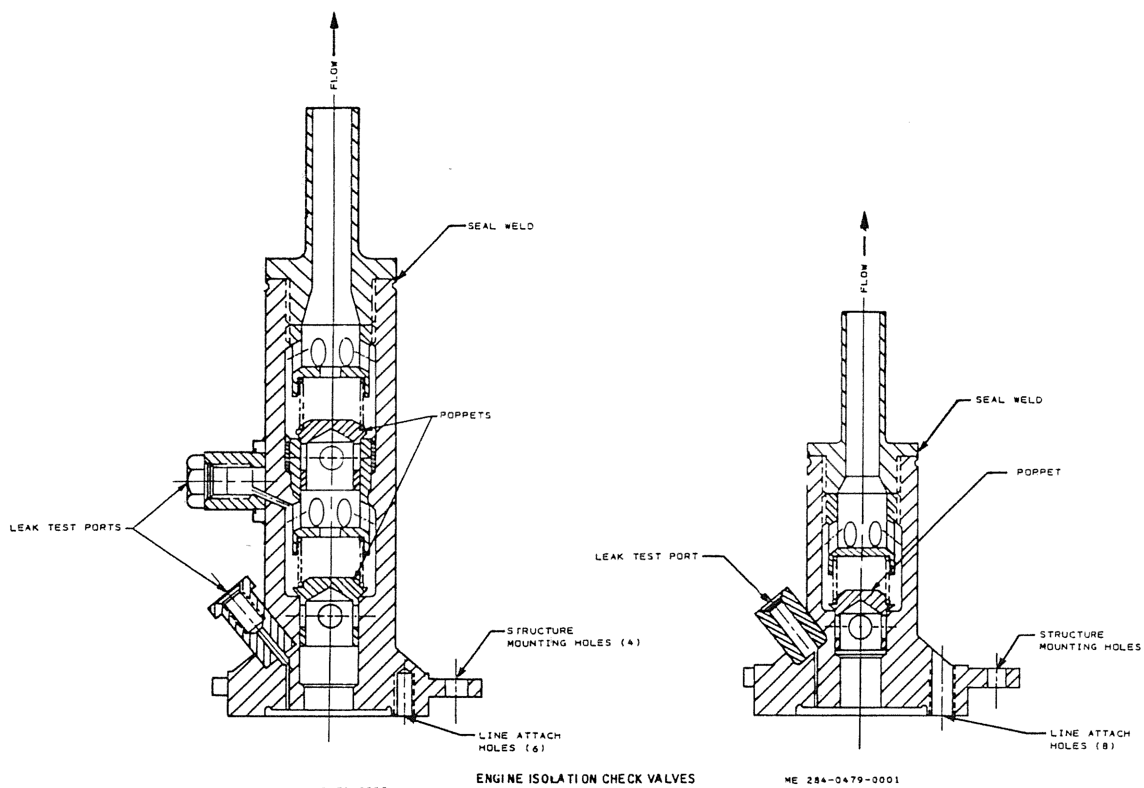
This SB describes the following MPS valves located in the orbiter:

- Engine isolation check valves (p. 3.3-2)
- LO2 bleed check valve (p. 3.3-3)
- Helium check valves (p. 3.3-4)
- Manifold relief valves (p. 3.3-5)
- Helium supply relief valves (p. 3.3-7)
- Fill and drain valves (p. 3.3-9)
- Prevalves (p. 3.3-12)
- Manifold repress regulators (p. 3.3-15)
- Helium supply regulators (p. 3.3-16)
- MPS ball valves (p. 3.3-18)
- Feedline relief shutoff valves (p. 3.3-20)
- Flow control valves (p. 3.3-21)
- Helium control valves (p. 3.3-23)
- Helium supply valves (p. 3.3-24)
- Control valves for pneumatic-operated valves (p. 3.3-25)
- Orbiter ground disconnects (p. 3.3-26)

3.3.2 Valve Descriptions

3.3.2.1 Engine Isolation Check Valves

A schematic of the engine isolation check valves appears in figure 3.3-1. These valves provide flow isolation for the center, left, and right SSME GO2 and GH2 pressurization lines. When upstream pressure exceeds downstream pressure, the in-line, spring-loaded closed poppet will be pushed open. If the downstream pressure is higher, the check valve will be spring-loaded closed.



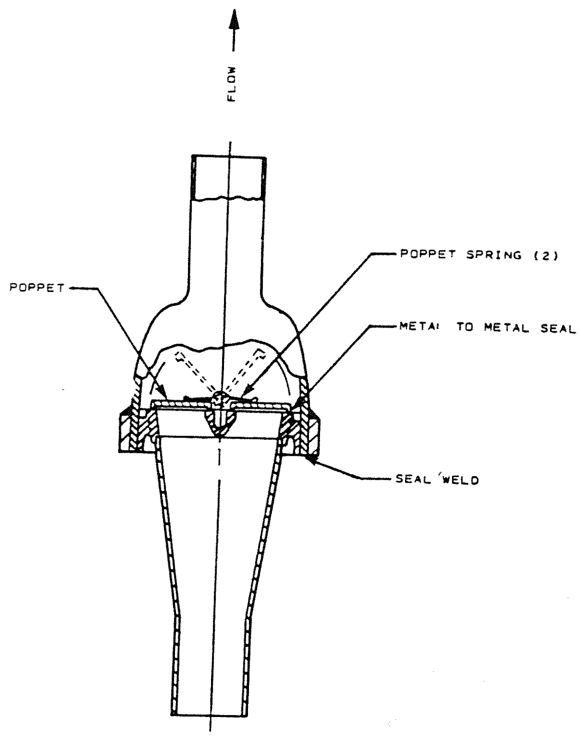
ENGINE ISOLATION CHECK VALVES

DASH NO	USE
0001	1. C.L.R. GO_2 PRESS ISOLATION CHECK VALVE
0002	2. C.L.R. GH_2 PRESS ISOLATION CHECK VALVE

Figure 3.3-1.- Engine isolation check valves.

3.3.2.2 LO2 Bleed Check Valves

A schematic of the LO2 bleed check valves is given in figure 3.3-2. These valves provide flow isolation for the center, left, and right SSME LO2 bleed lines and are typical in-line, spring-loaded-closed check valves.



LO_2 BLEED CHECK VALVES
ME284-0515-0001

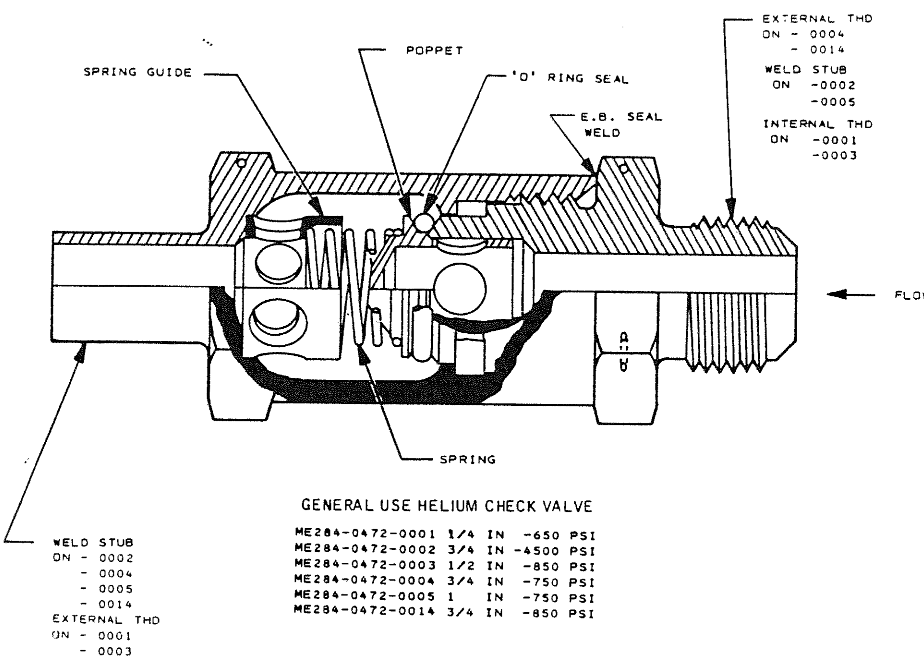
DASH NO	USE
0001	1. C,L,R, LO_2 BLEED CHECK VALVE

Figure 3.3-2.- LO_2 bleed check valves.

3.3.2.3 General Use Helium Check Valves

A typical schematic of the general use helium check valves used in the orbiter is provided in figure 3.3-3. These valves are typical in-line, spring-loaded-closed check valves which provide flow isolation for the following lines:

- GO2 and GH2 pressurization manifold repress
- LH2 recirc manifold repress
- Center, left, and right SSME helium supply and regulator outlets
- Pneumatic helium supply, isolation, and regulator outlet
- Helium interconnects
- LO2 and LH2 feedline manifold repress
- LH2 feed manifold RTLS press
- LO2 and LH2 tank prepress

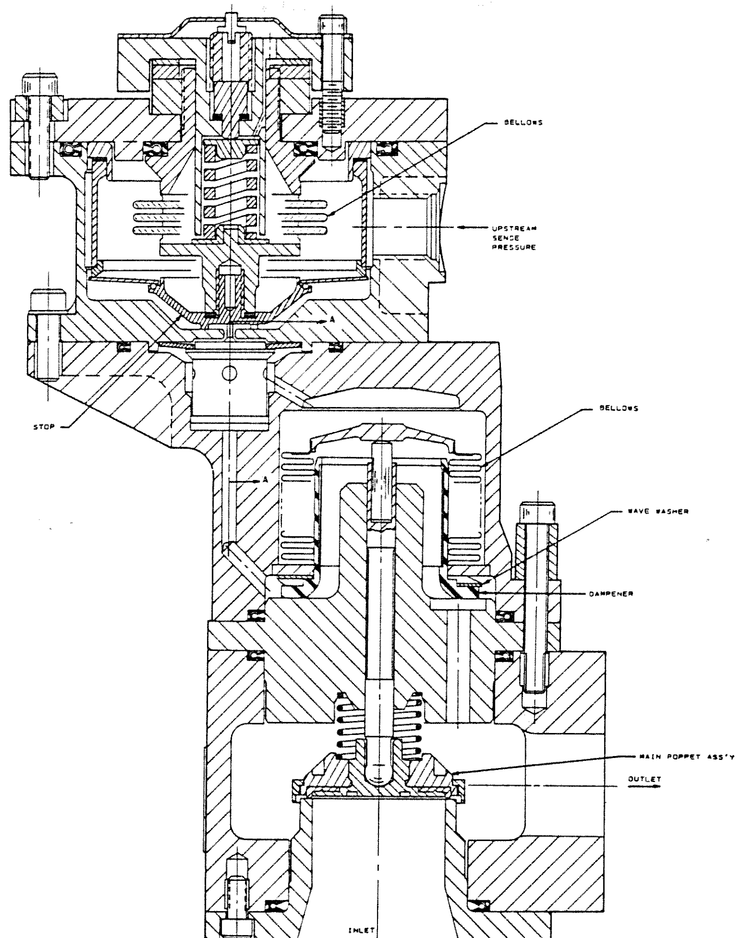


DASH NO	USE
0001	1. GO_2 PRESS MANIF REPRESS CHECK VALVE
	2. LH_2 RECIRC MANIF REPRESS CHECK VALVE
	3. GH_2 PRESS MANIF REPRESS CHECK VALVE
0002	1. C,L,R, HELIUM SUPPLY CHECK VALVE
	2. PNEU HE SUPPLY CHECK VALVE
	3. HE INTERCONNECT CHECK VALVES
0003	1. LO_2 FEED MANIF REPRESS CHECK VALVE
	2. PNEU HE ISOLATION CHECK VALVE
0004	PNEU HE REG OUT CHECK VALVE
	2. LH_2 FEED MANIF RTLS PRESS CHECK VALVE
	LH_2 FEED MANIF (NOM) REPRESS CHECK VALVE
0005	1. LO_2 TANK PREPRESS CHECK VALVE
	2. LH_2 TANK PREPRESS CHECK VALVE
0014	C.L.R ENGINE HELIUM REG OUT CHECK VALVE

Figure 3.3-3.- General use helium check valve.

3.3.2.4 Manifold Relief Valves

A schematic of the manifold relief valves is shown in figure 3.3-4. The LO2 and LH2 manifold relief valves have 1-inch inside diameter inlets and outlets and relieve at 190/220 psig and 40/45 psig, respectively. The main poppet of these valves is normally held closed by upstream pressure on a bellows cylinder head. When the upstream sensed inlet pressure reaches the relief setting, the pilot poppet opens, vents down the pressure on the bellows, and allows the inlet pressure to push open the main poppet. Once the manifold pressure drops below the pilot control setting, the manifold pressure is allowed to enter the closing chamber and the main poppet closes. The recirculation manifold relief valve is an in-line, spring-loaded valve (0.5 inch inside diameter) which relieves at 25 psid.



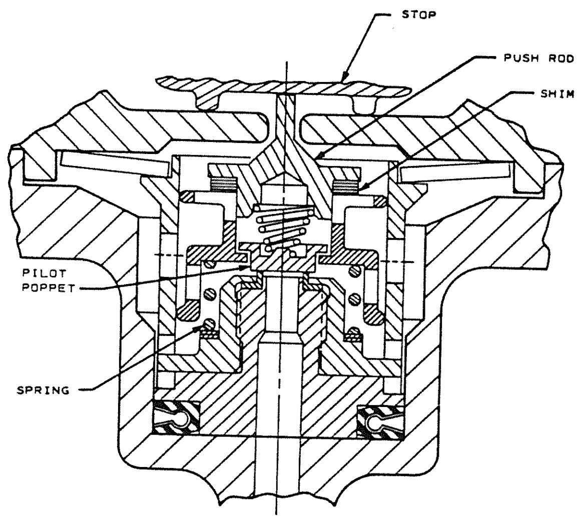
MANIFOLD RELIEF VALVES

MC284-0501-0001 1 IN. 190/220 PSIG RLF

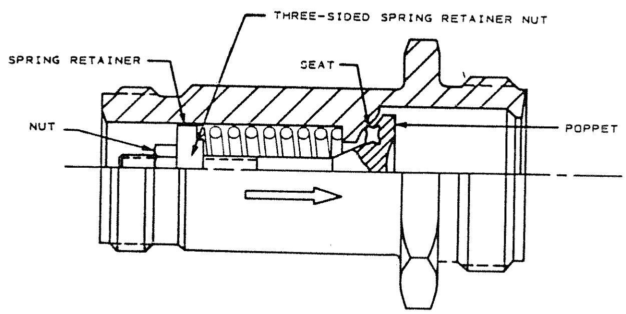
MC284-0501-0002 1 IN. 40/45 PSIG RLF

DASH NO	USE
0001	1. LO_2 FEED MANIF RELIEF VALVE
0002	1. LH_2 FEED MANIF RELIEF VALVE

Figure 3.3-4.- Manifold relief valves (page 1 of 2).



SECTION A-A
PILOT POPPET ASSY



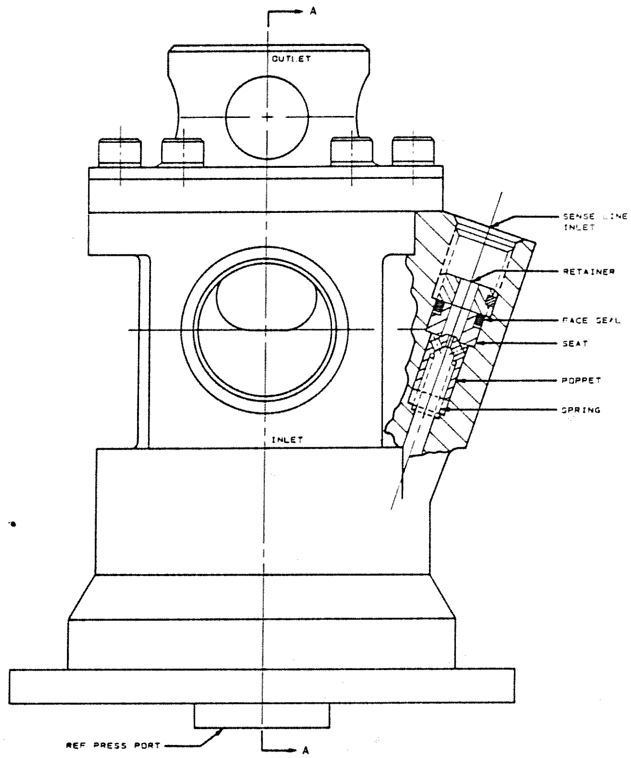
RECIRC MANIFOLD RLF VALVES
ME284-0474-0002 1/2 IN 25 PSI CRACK

DASH NO	USE
0002	1. LH_2 RECIRC MANIFOLD RELIEF VALVE

Figure 3.3-4.- Manifold relief valves (page 2 of 2).

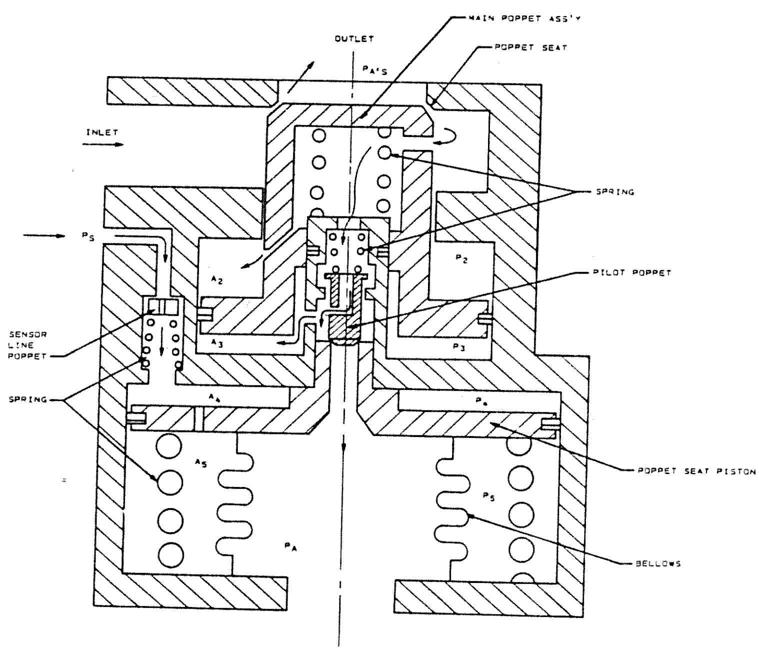
3.3.2.5 Helium Supply Relief Valves

A schematic of the helium supply relief valves is provided in figure 3.3-5. The engine and pneumatic helium supply relief valves open at 800/850 psig and reseat at 785 psig. The main poppet of this valve is normally spring-loaded closed, and the pressures on both sides of the main poppet piston are equal. When the upstream sensed inlet pressure reaches the relief setting, the sense line poppet opens and the pilot poppet seat piston is pushed downward by the sensed pressure away from the pilot poppet, thus allowing the backside of the main poppet piston to be vented to ambient and the inlet pressure to force open the main poppet.



HELIUM SUPPLY RELIEF VALVE
MC284-0398-0004

DASH NO	USE
0004	1. C. L. R. ENGINE HELIUM SUPPLY RELIEF VALVE 2. PNEU HELIUM SUPPLY RELIEF VALVE



SIMPLIFIED DIAGRAM OF
RELIEF VALVE OPERATION

Figure 3.3-5.- Helium supply relief valve (page 1 of 2).

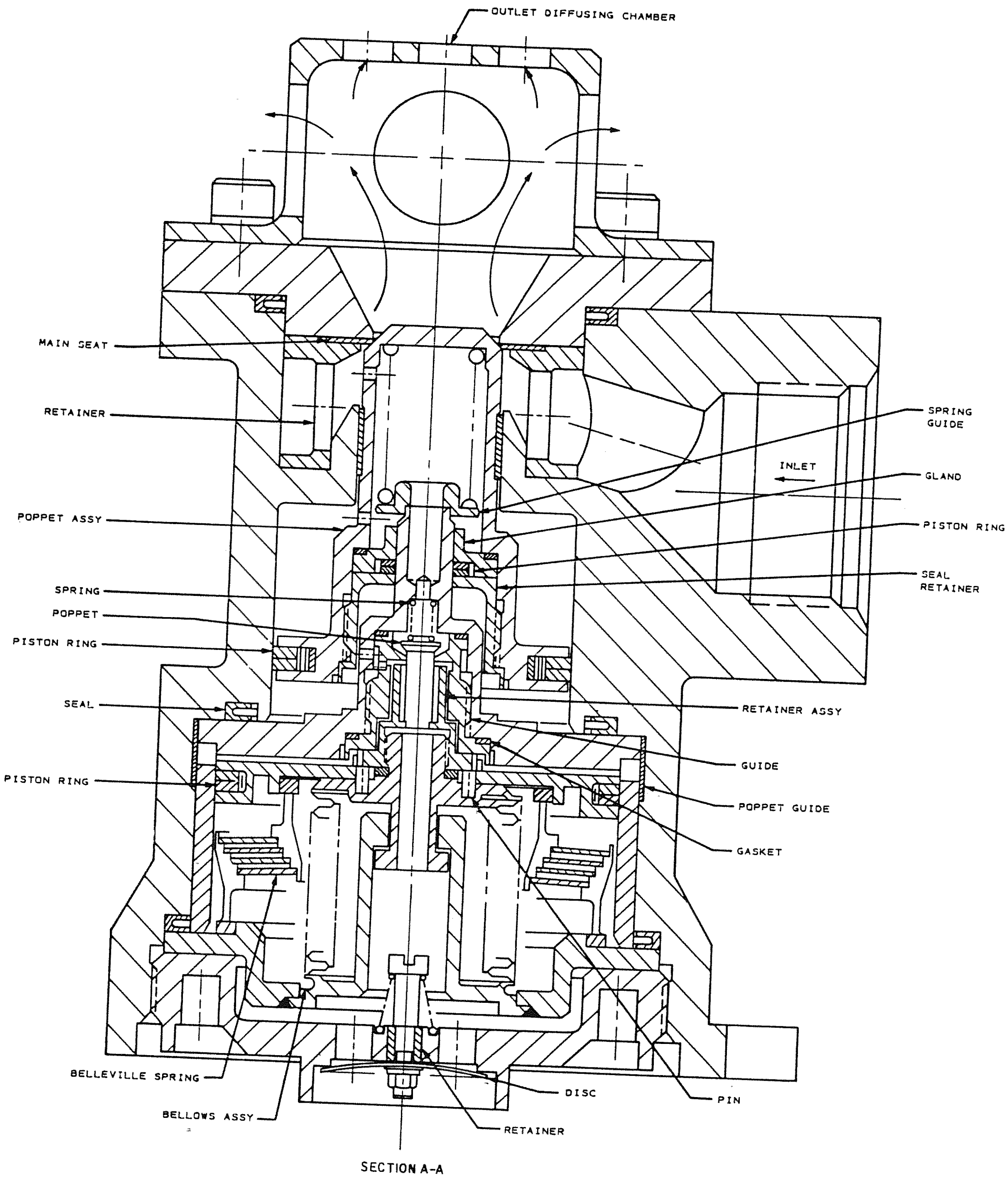
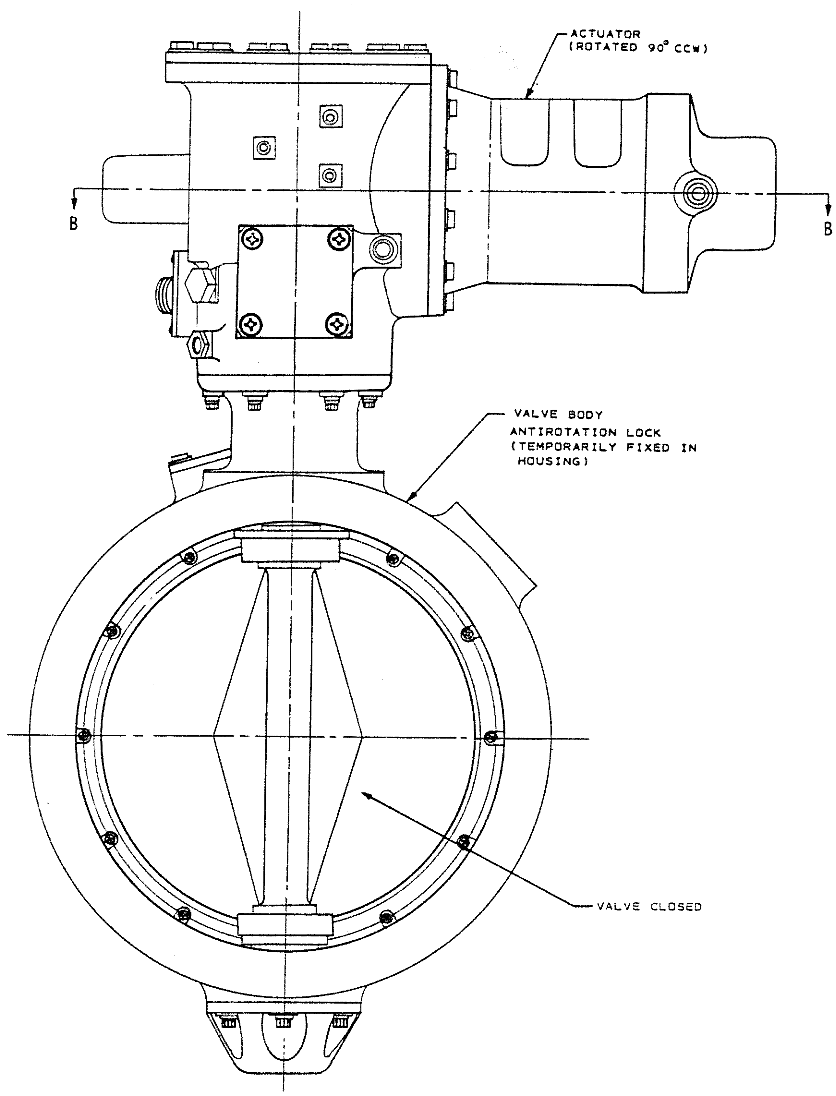


Figure 3.3-5.- Helium supply relief valve (page 2 of 2).

3.3.2.6 Fill And Drain Valves

A schematic of the LO2 and LH2 inboard and outboard fill and drain valves appears in figure 3.3-6. The fill and drain valves are two-position, pneumatically-actuated valves located in the 8-inch diameter fill and drain lines. The valves are open during propellant loading, LH2 dump (LH2 valves only), and vacuum inerting.



FILL AND DRAIN VAtVE
MC284-0397-0007
0008
0009

DASH NO.	USE
0007	1. LO_2 OUTBD FILL/DRAIN VALVE
0008	1. LO_2 INBD FILL/DRAIN VALVE
0009	1. LH_2 INBD/OUTBD FILL/DRAIN VALVE

Figure 3.3-6.- Fill and drain valves (page 1 of 2).

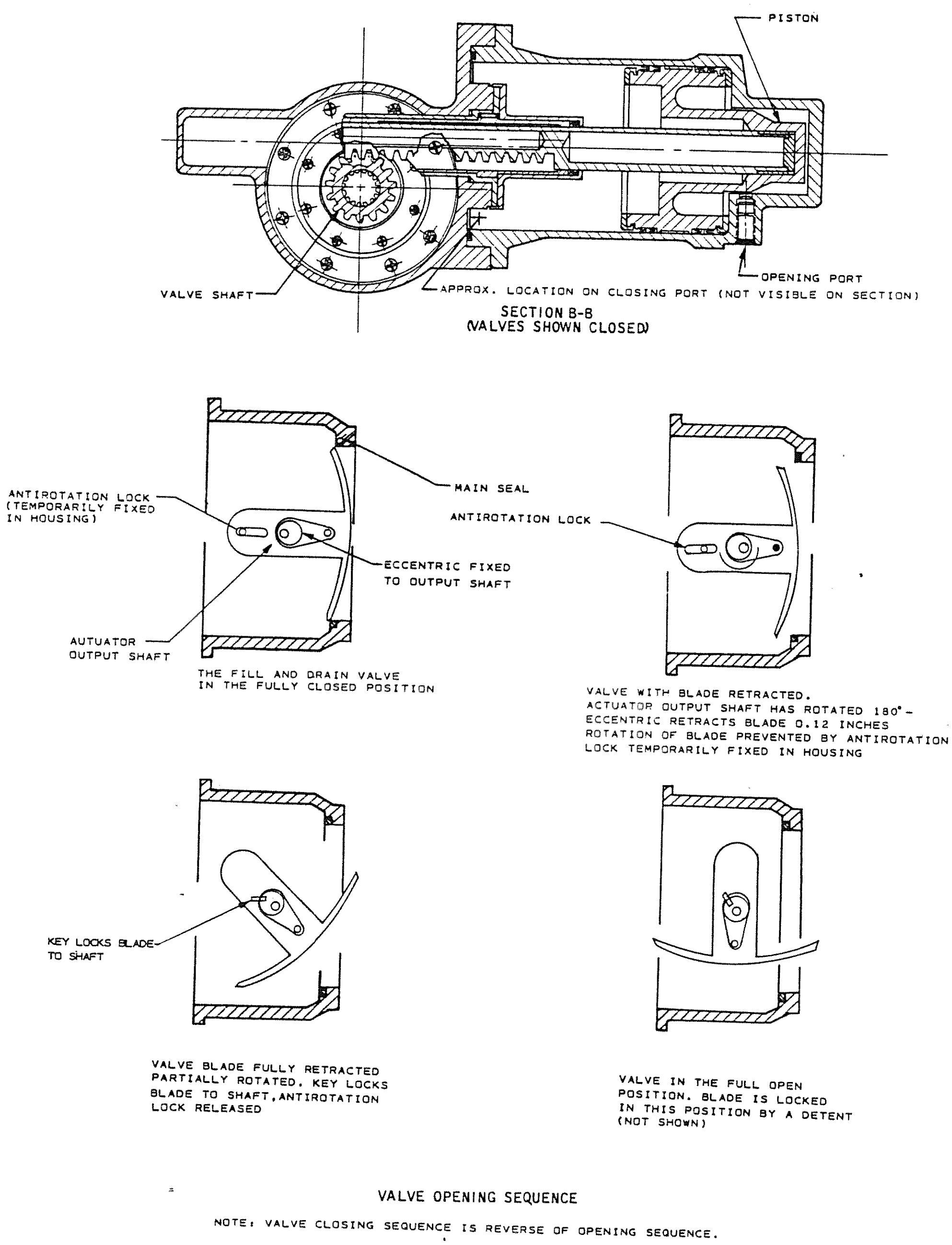
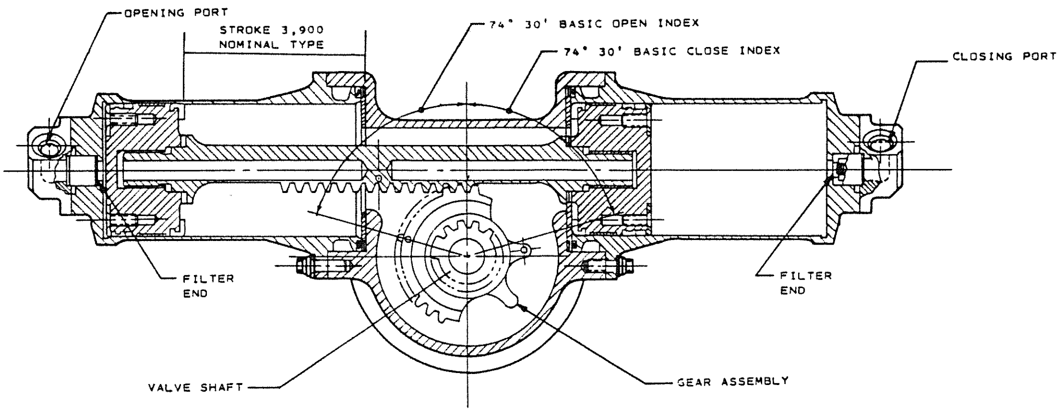


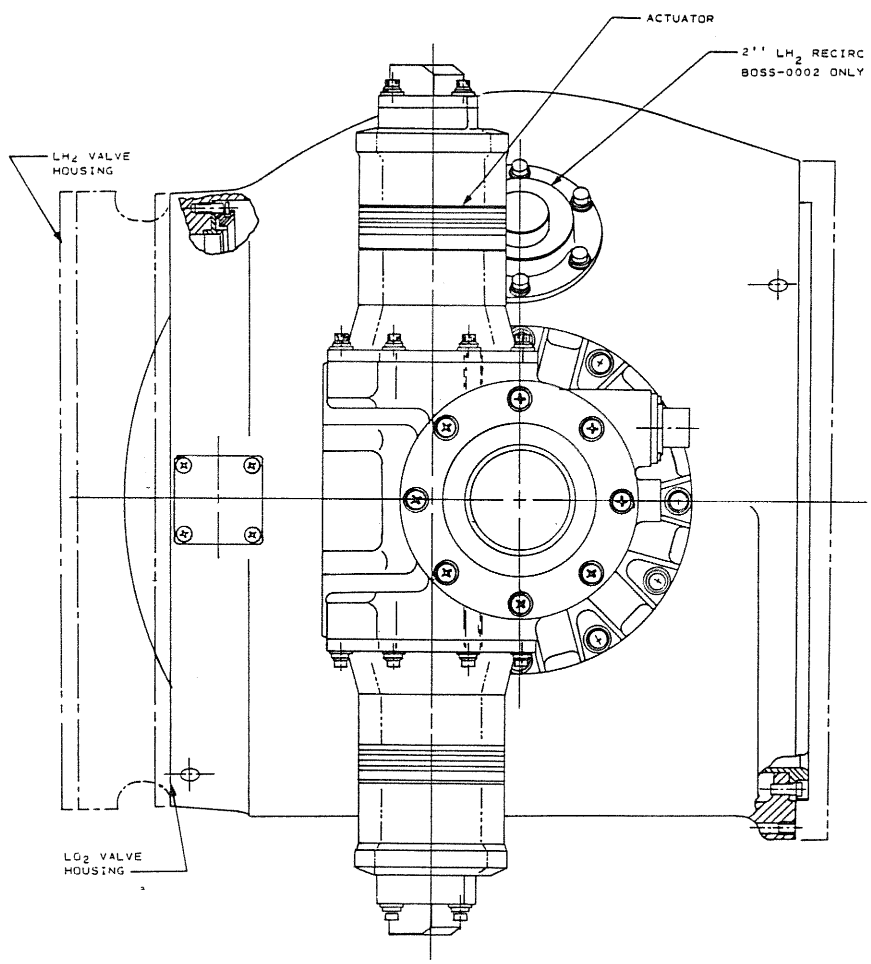
Figure 3.3-6.- Fill and drain valves (page 2 of 2).

3.3.2.7 Prevalves

A schematic of the LO2 and LH2 prevalves is shown in figure 3.3-7. The prevalves are two-position, pneumatically-actuated valves in the 12-inch diameter LO2 and LH2 engine feedlines (six valves total). The prevalves have two means of internal relief; the main visor and a visor bypass poppet valve. With closed position solenoid actuator vented, the main visor will crack and reset at 15 psid, maximum. The poppet will crack and reseat at 15 to 50 psid under cryo conditions. Antislam protection is provided for the valve during opening or closing by pressurizing the opposite actuator piston force, thus slowing valve travel. Actuation pressure pushes the spring-loaded antislam poppet open, allowing helium to pressurize the opposite piston face to a pressure less than the actuation pressure. When the piston has moved most of the way to the new position, it pushes the antislam poppet closed and the opposite piston face vents down through the nonenergized actuator solenoid valve.



SECTION THROUGH ACTUATOR
(VALVE SHOWN CLOSED)



PREVALVES
MC284-0396-0003
0004

DASH NO.	USE
0003	1. E1,E2,E3 LO_2 PREVALVE
0004	1. E1,E2,E3 LH_2 PREVALVE

Figure 3.3-7.- Prevalves (page 1 of 3).

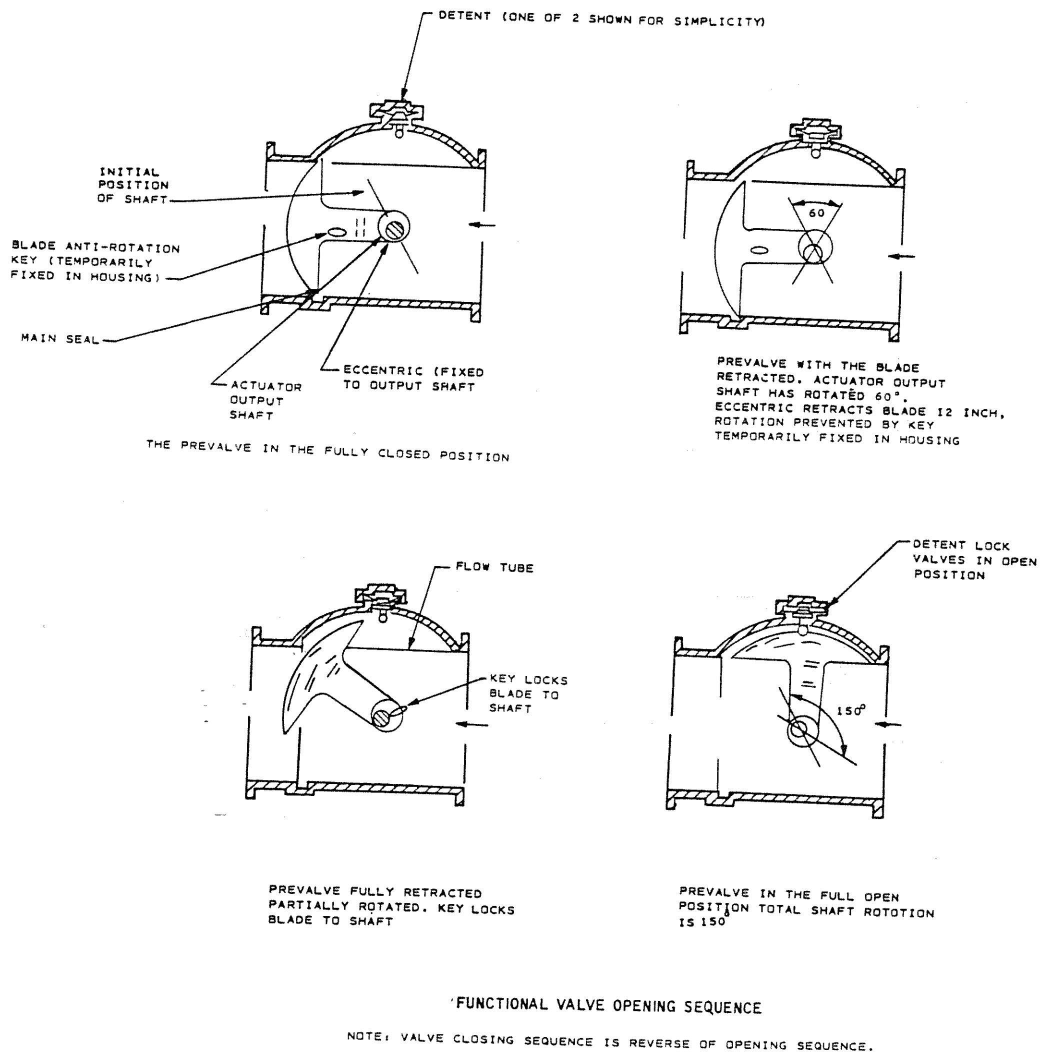


Figure 3.3-7.- Prevalves (page 2 of 3).

COMPONENT: PREVALVE, SHUTOFF, PROPELLANT

(MC284-0396)
ANTISLAM ACTUATOR

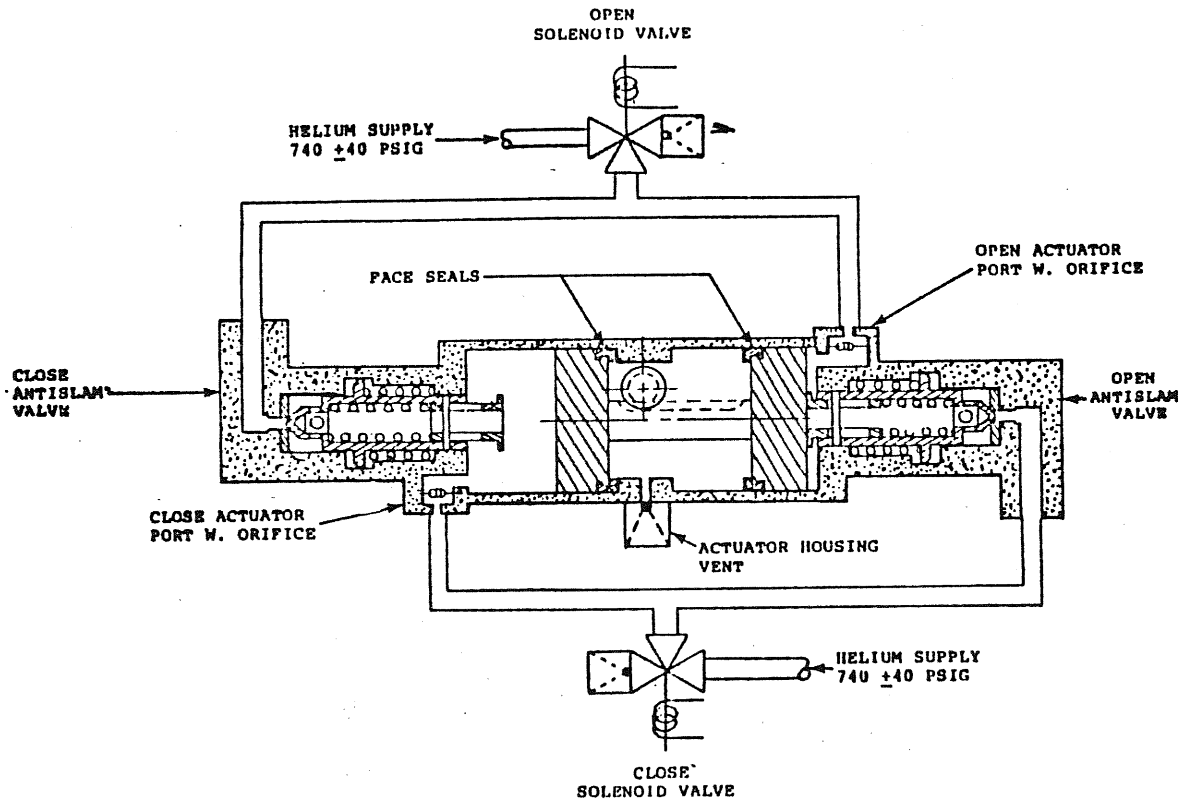
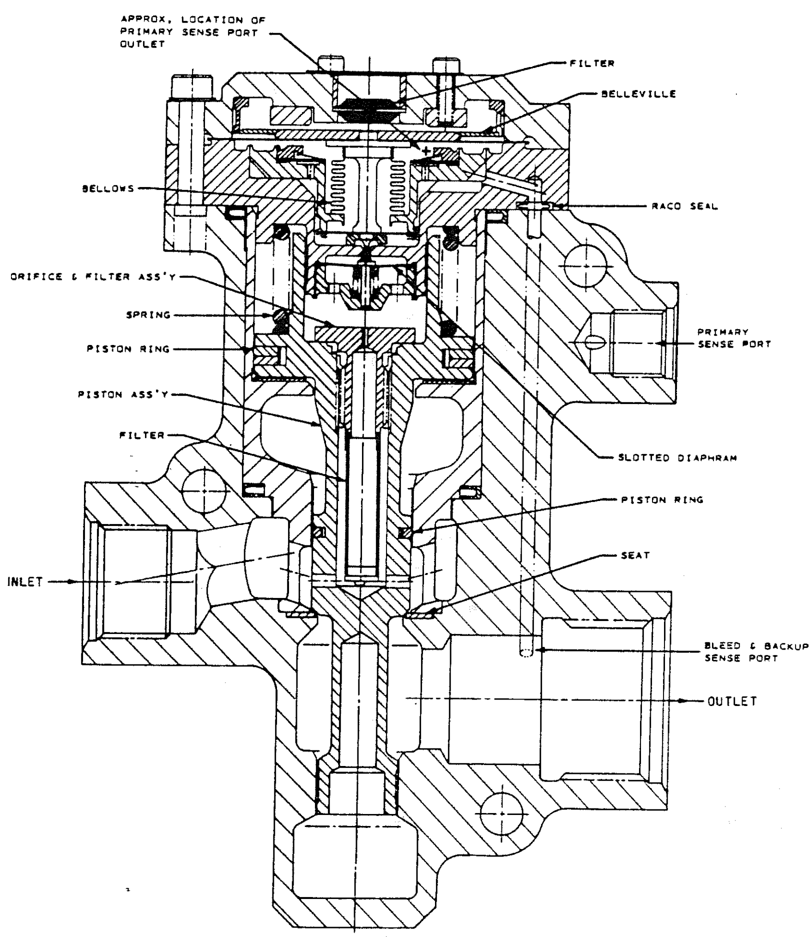


Figure 3.3-7.- Prevalves (page 3 of 3).

3.3.2.8 Manifold Repress Regulators

A schematic of the LO2 and LH2 manifold repress regulators is given in figure 3.3-8. These regulators operate at 20/25 psig. When the sensed regulator outlet pressure is below the regulator setting, the pressure sensing bellows moves the pilot poppet to the open position, the upper main poppet cavity pressure is vented to the outlet pressure, and the higher inlet pressure on the lower main poppet face will force the main poppet open. When the sensed outlet pressure is above the regulator setting, the bellows moves the pilot poppet to the closed position, the pressure on both sides of the main poppet piston equalize to the inlet pressure, and the main poppet is spring-loaded closed.



MANIFOLD REPRESS REGULATOR

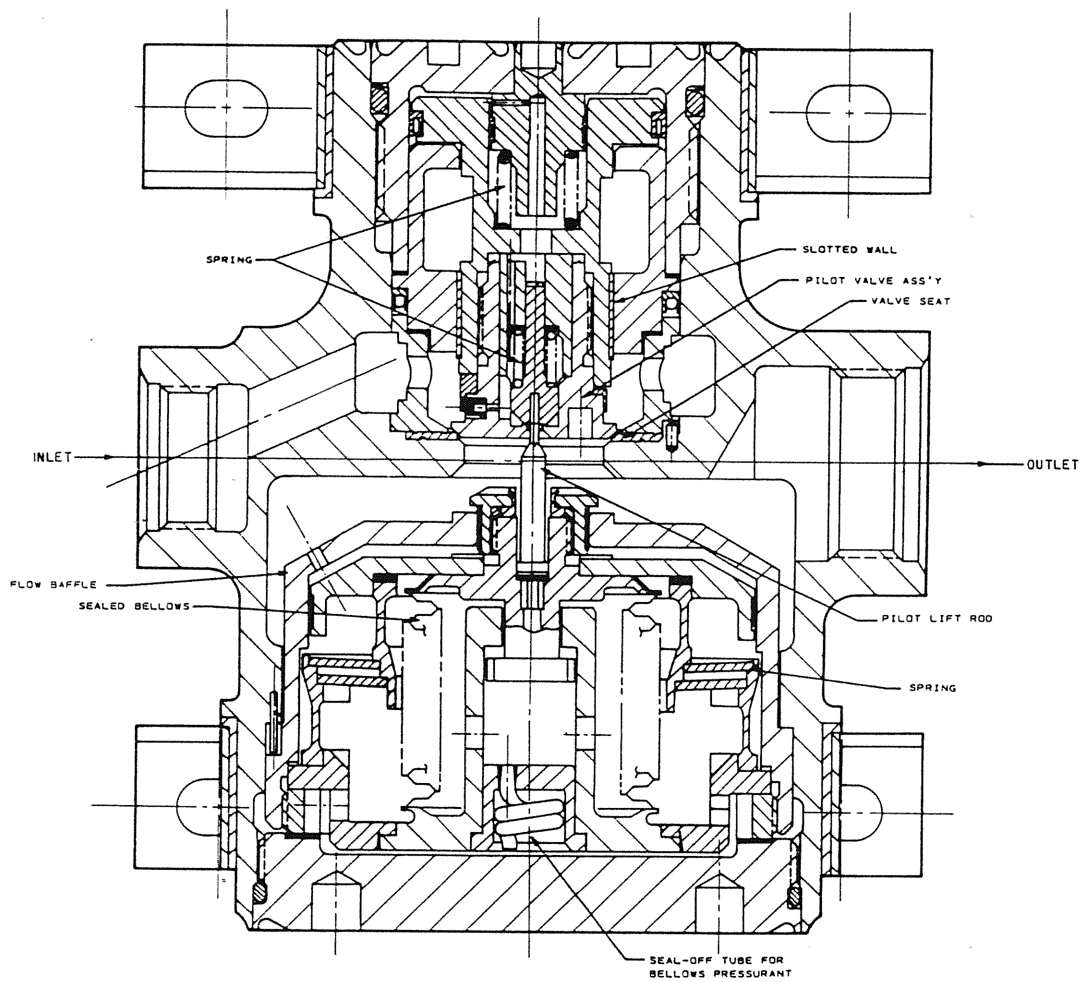
MC284-0399-0002

DASH NO.	USE
0002	1. LO_2 MANIF REPRESS REGULATOR 2. LH_2 MANIF REPRESS REGULATOR

Figure 3.3-8.- Manifold repress regulator.

3.3.2.9 Helium Supply Regulators

A schematic of the helium supply regulators (i.e., the six SSME helium regulators and the pneumatic helium supply regulator) appears in figure 3.3-9. These operate at 715/770 psig and lockup at 785 psig, minimum. When the regulator outlet pressure drops below the regulator setting, the outlet compartment bellows will expand and the pilot lift rod will push open the pilot poppet. This allows the cavity above the main poppet to bleed down to the outlet pressure and the main poppet is spring-loaded open. When the outlet pressure rises above the regulator setting, the bellows is compressed, the lift rod is retracted, and the pilot poppet is spring-loaded closed. This allows the cavity above the main poppet to repressurize to the higher inlet pressure and the main poppet is pushed closed. The regulator main poppet will go full open if the inlet pressure drops below 900 psia.



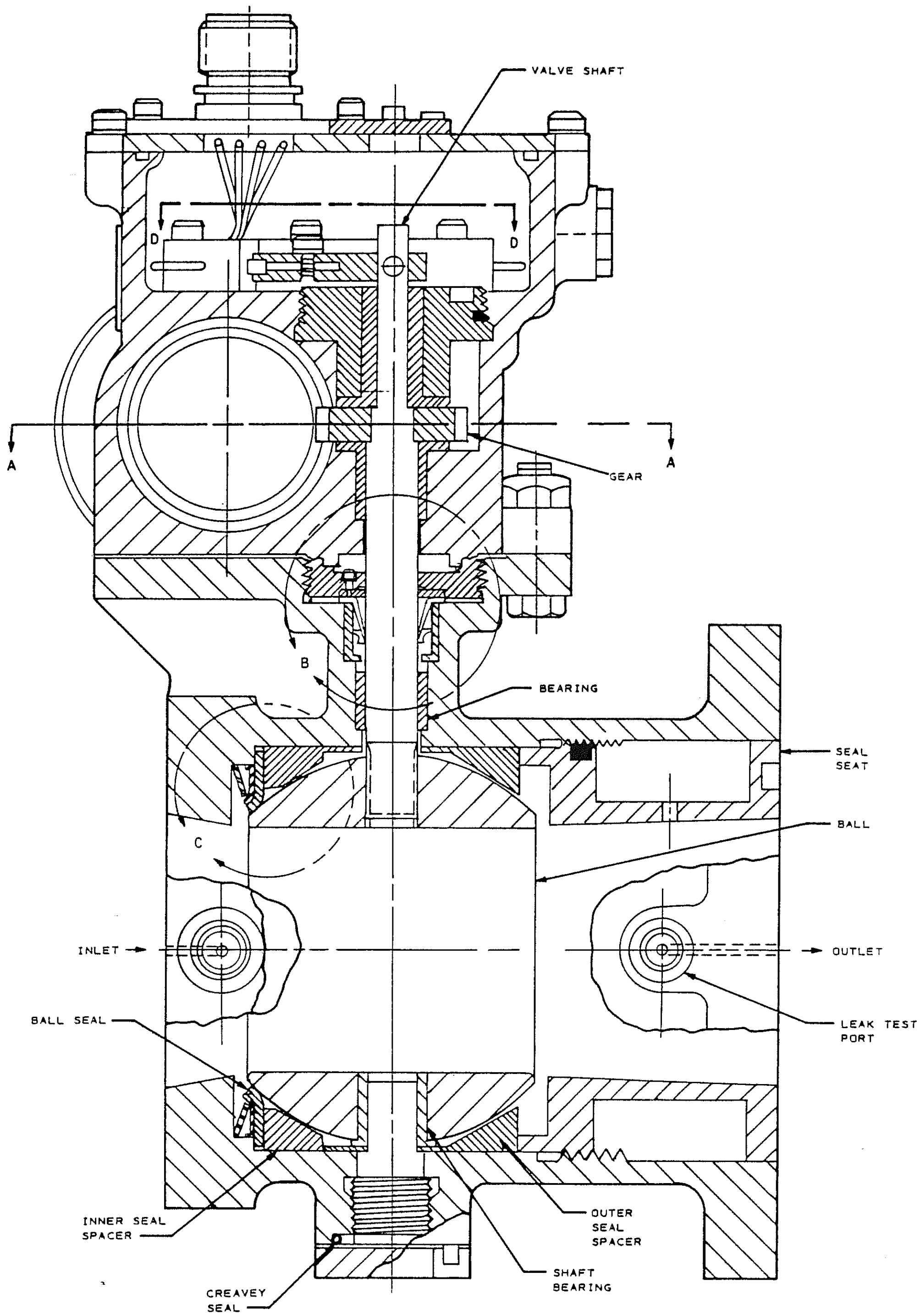
HELIUM SUPPLY REGULATORS
MC284-0533-0004

DASH NO.	USE
0004	1. E1,E2,E3 HELIUM SUPPLY REGULATOR 2. PNEU VALVE HELIUM SUPPLY REGULATOR

Figure 3.3-9.- Helium supply regulators.

3.3.2.10 MPS Ball Valves

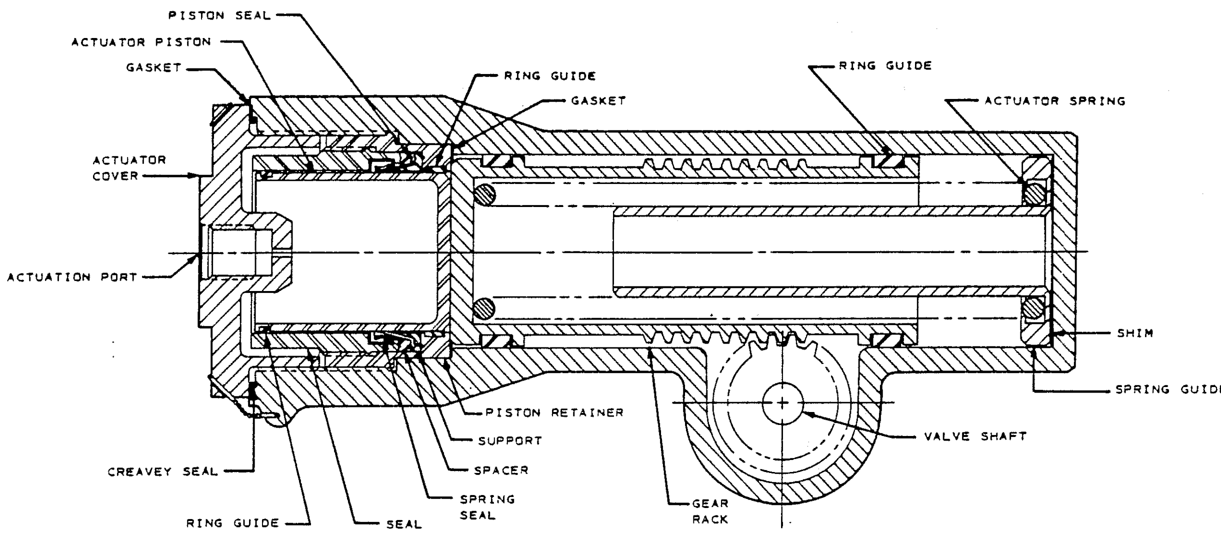
A typical schematic of the MPS pneumatically actuated ball valves is shown in figure 3.3-10. The LH_2 topping valves and center, left, and right LH_2 recirculation pump valves are 2-inch diameter, normally closed valves. The LO2 POGO valves are 2-inch diameter, normally open valves. The POGO valve is designed to relieve past the main ball seal to prevent pressure buildup between the pogo valve and the LO_2 bleed valve when both valves are closed. The valve relieves into the LO_2 manifold. The LH_2 RTLS dump valves and high point bleed valves are normally closed, 1.5-inch valves. The LH_2 inboard RTLS dump valve has an integral relief valve to prevent the buildup of pressure between the inboard and outboard RTLS valves when LH_2 is trapped between the valves. The LO_2 inboard bleed valve is a normally open, 1.5-inch diameter valve with integral relief to prevent buildup of pressure between the valve outlet and the LO_2 bleed disconnect. These valves utilize a geared (rack and pinion) actuator piston and valve shaft, with a spring-loaded return following removal of actuation pressure.



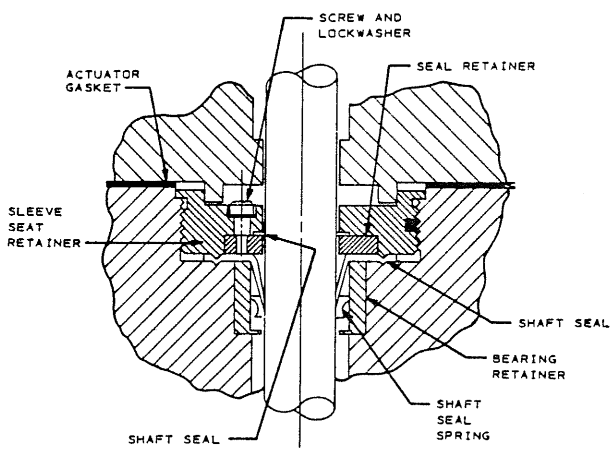
BALL VALVES

MC284-0395-0011	2 IN NC
	-0012	2 IN N.O.
	-0013	1 1/2 IN NC
	-0014	1 1/2 IN NC
	-0015	1 1/2 IN N.O.

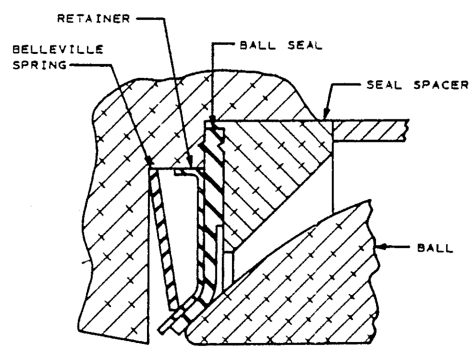
Figure 3.3-10.- MPS ball valve (page 1 of 2).



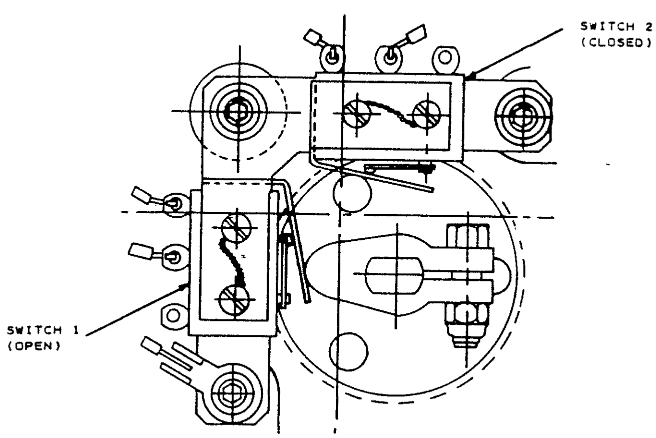
VIEW A-A
(ROTATED 90° CCW)



VIEW B



VIEW C



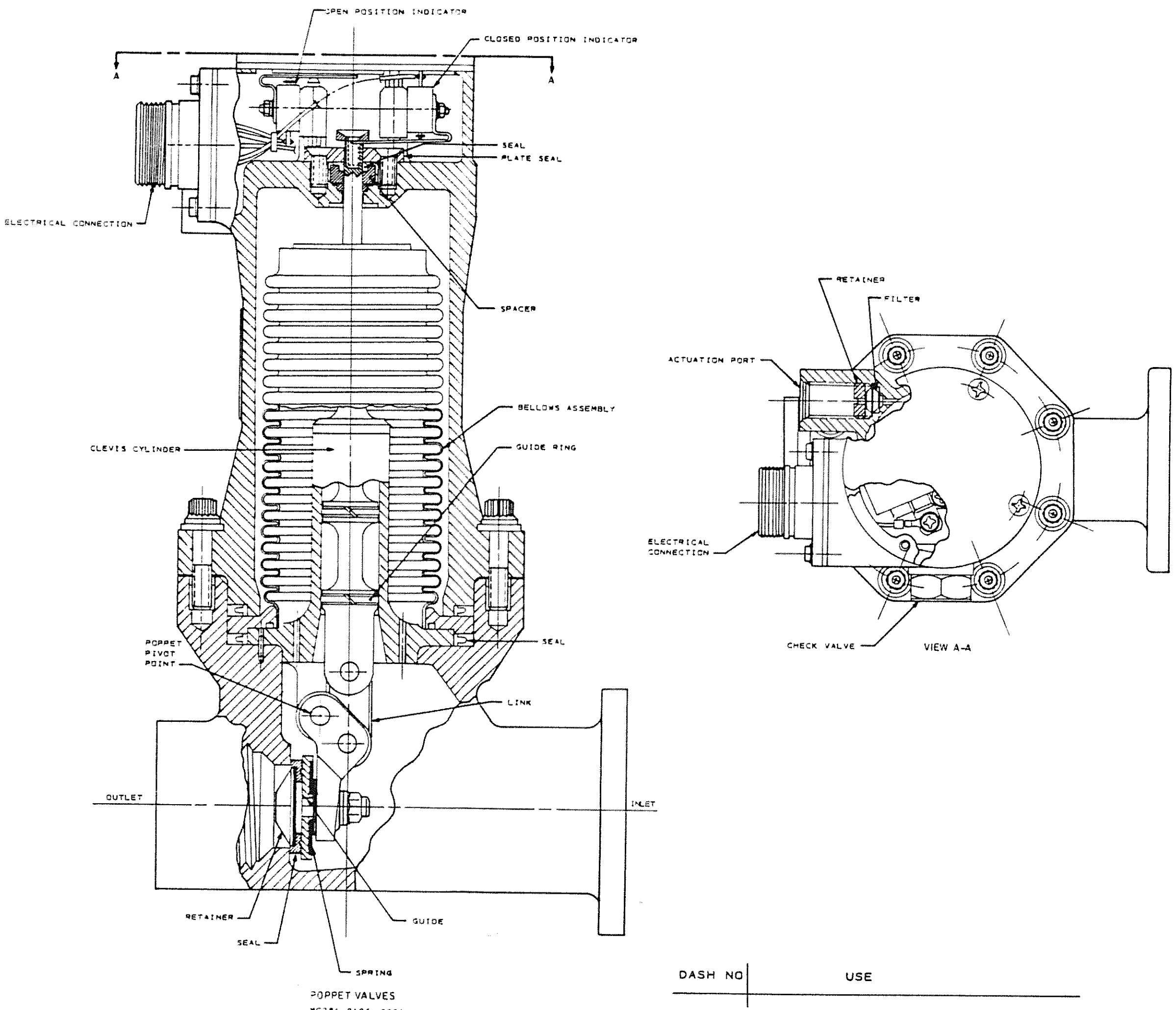
VIEW D-D
(TYPICAL SWITCH LAYOUT)

DASH NO	USE
0001	1. LH_2 REPLENISH (TOPPING) VALVE 2. C,L,R LH_2 RECIRC PUMP VALVE
0002	1. LO_2 POGO ACCUM RECIRC VALVES NO 1 AND NO 2
0003	1. LO_2 RTLS INBD DUMP VALVE 2. MPS HIGH POINT BLEED VALVE
0004	1. LH_2 RTLS OUTBD DUMP VALVE
0005	1. LO_2 INBD BLEED VALVE

Figure 3.3-10.- MPS ball valve (page 2 of 2).

3.3.2.11 Feedline Relief Shutoff Valves

A schematic of the feedline relief shutoff valves is provided in figure 3.3-11. These valves are normally open, pneumatically-actuated poppet valves. The valve poppet is mechanically linked to a bellows assembly which is compressed by actuation pressure (thus opening the poppet) and expands when actuation pressure is vented (closing the poppet).



DASH NO	USE
0001	1. LO_2 FEEDLINE RELIEF S/OFF VALVE 2. LH_2 FEEDLINE RELIEF S/OFF VALVE

Figure 3.3-11.- Feedline relief shutoff valves.

3.3.2.12 Flow Control Valves

A schematic of the original ET pressurant flow control valves appears in figure 3.3-12a. The center, left, and right SSME GO_2 and GH_2 flow control valves each have a bypass channel and a control channel equipped with a normally open solenoid valve. Each control valve is electrically controlled by its respective LO_2 or LH_2 ullage pressure electronics package; the valve is energized closed at the top of the control band, and deenergized at the bottom of the control band.

A schematic of the redesigned GH_2 flow control valve (STS-9 and subs) appears in figure 3.3-12b. This valve has two flow positions, and is spring-loaded to the high flow position unless solenoid-actuated to the low flow position. Valve control is the same as for present valves. On STS-26 and subs, the GO_2 flow control valve will be the same as the new GH_2 flow control valve except for the use of MONEL in the interior flow path to reduce the risk of partical induced ignition.

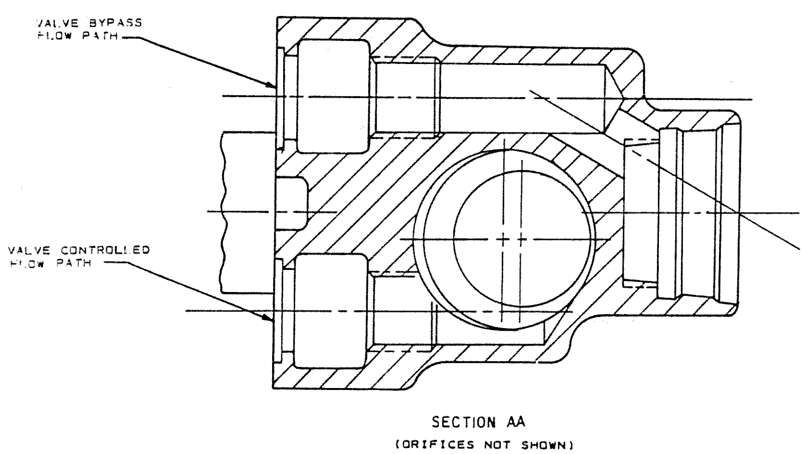
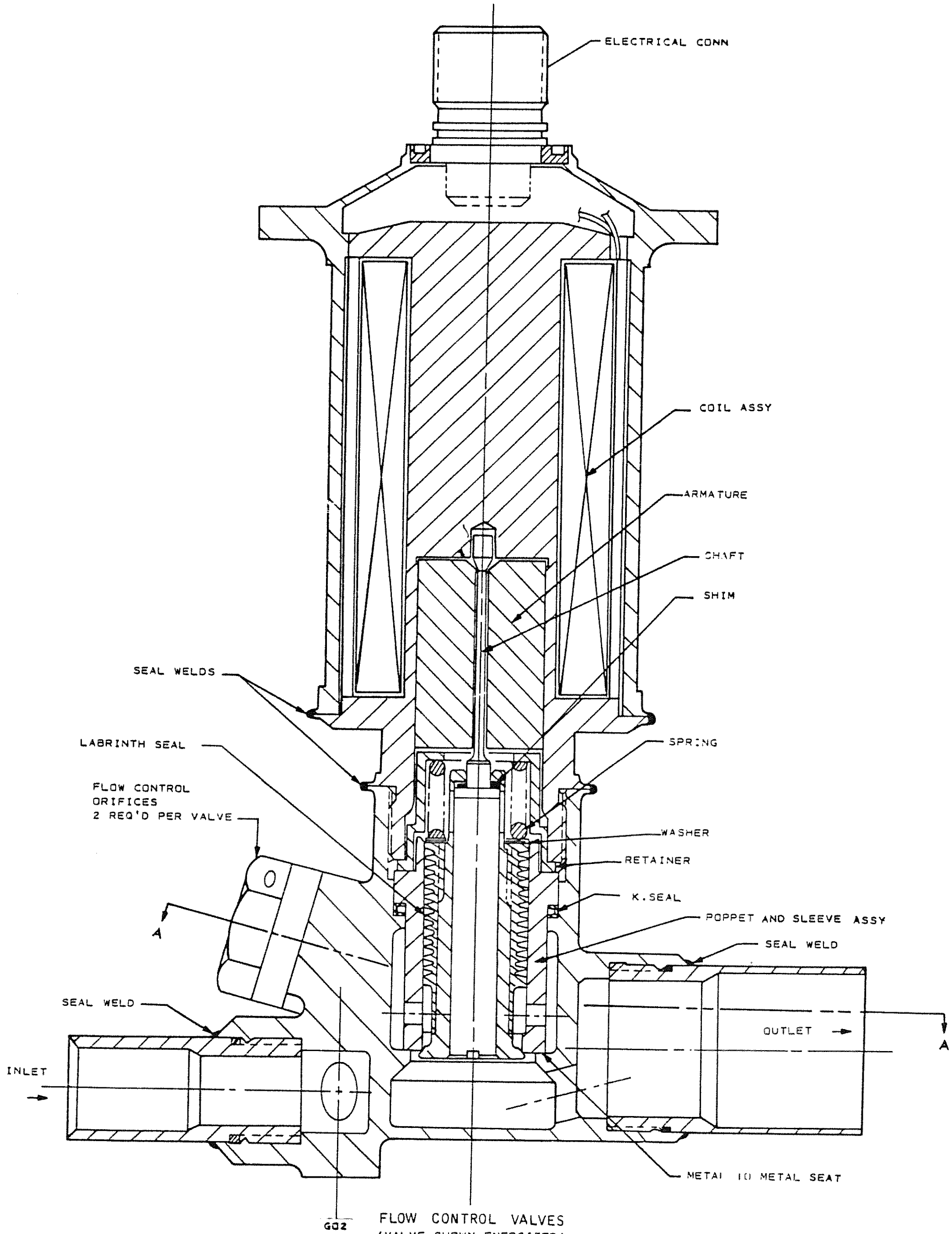


Figure 3.3-12a.- Flow control valves (page 1 of 2).



FLOW CONTROL VALVES
(VALVE SHOWN ENERGIZED)

MC280-0017-0001 4500 PSI
MC280-0017-0002 5500 PSI

DASH NO	USE
0001	1. C,L,R ENGINE GH2 PRESS FLOW CONTROL VALVE
0002	2. C,L,R ENGINE GO2 PRESS FLOW CONTROL VALVE

Figure 3.3-12a.- Flow control valves (page 2 of 2).

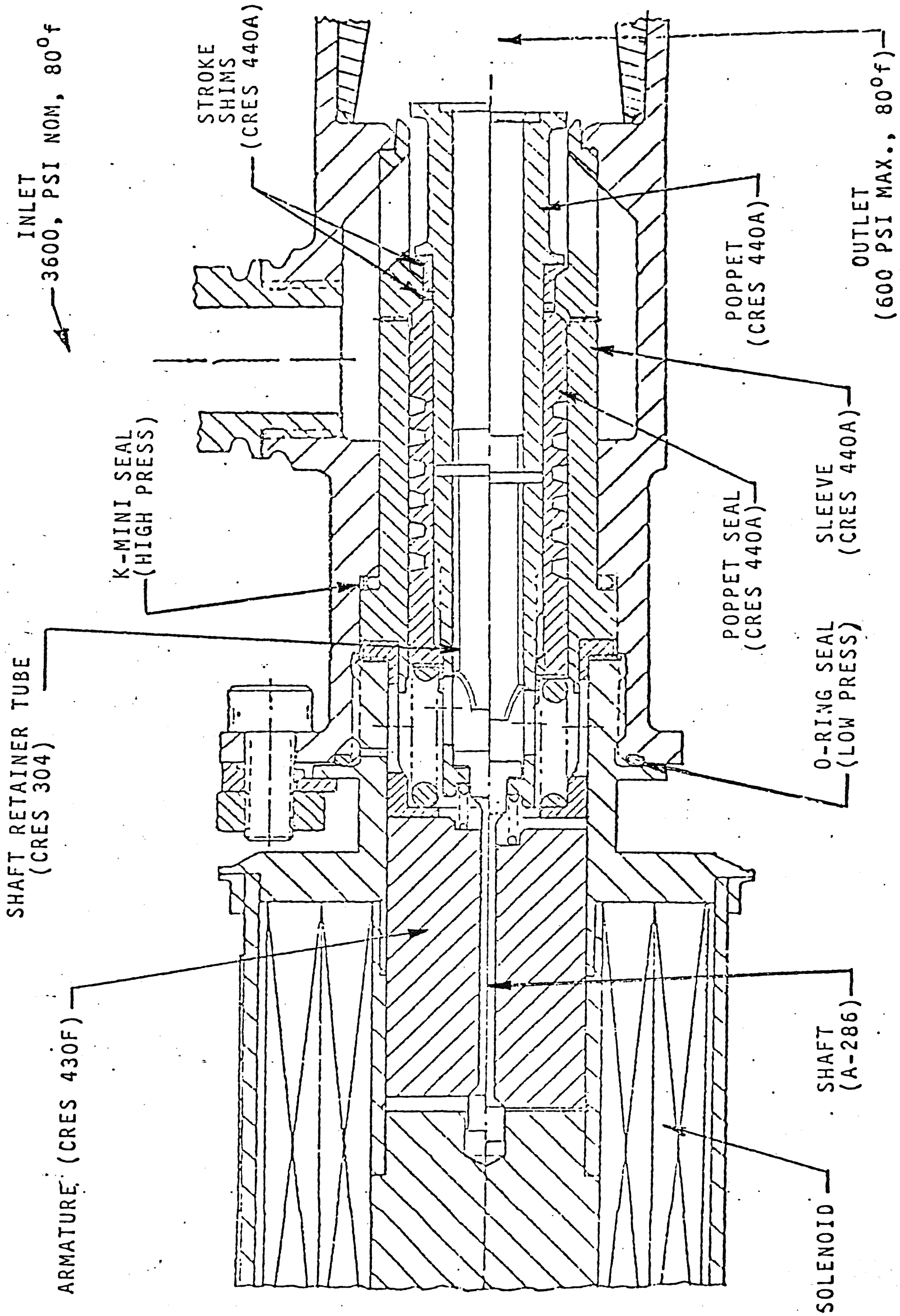
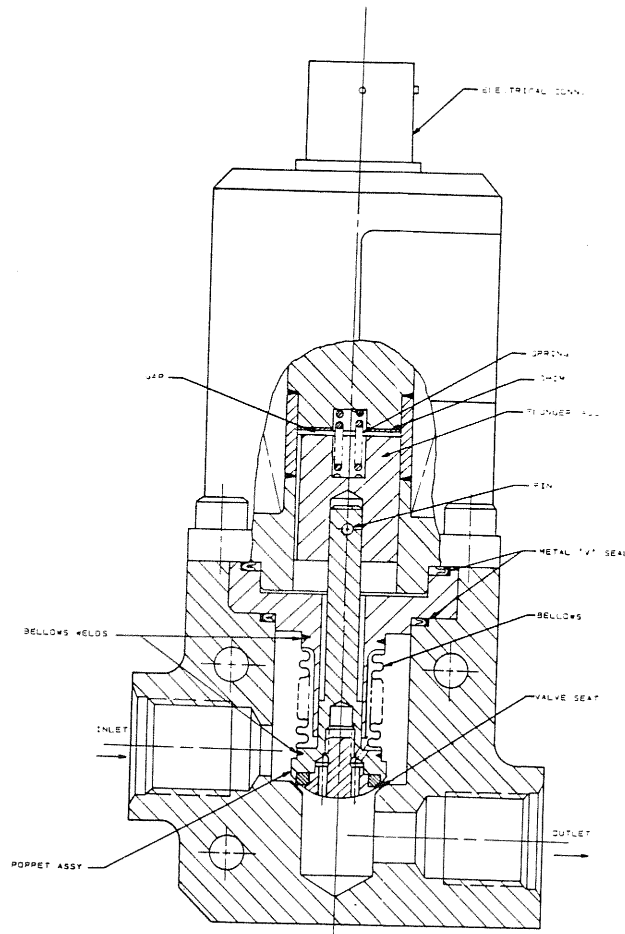


Figure 3.3-12b.- Redesigned GH2 flow control valve.

3.3.2.13 Helium Control Valves

A schematic of the helium control valves is presented in figure 3.3-13. These normally closed solenoid valves control flow for the following functions:

- LH2 feedline manifold RTLS repress
- GH2 pressurization line vent
- Helium supply blowdown
- Left SSME helium crossover
- LO2 and LH2 manifold repress



HELIUM CONTROL VALVES

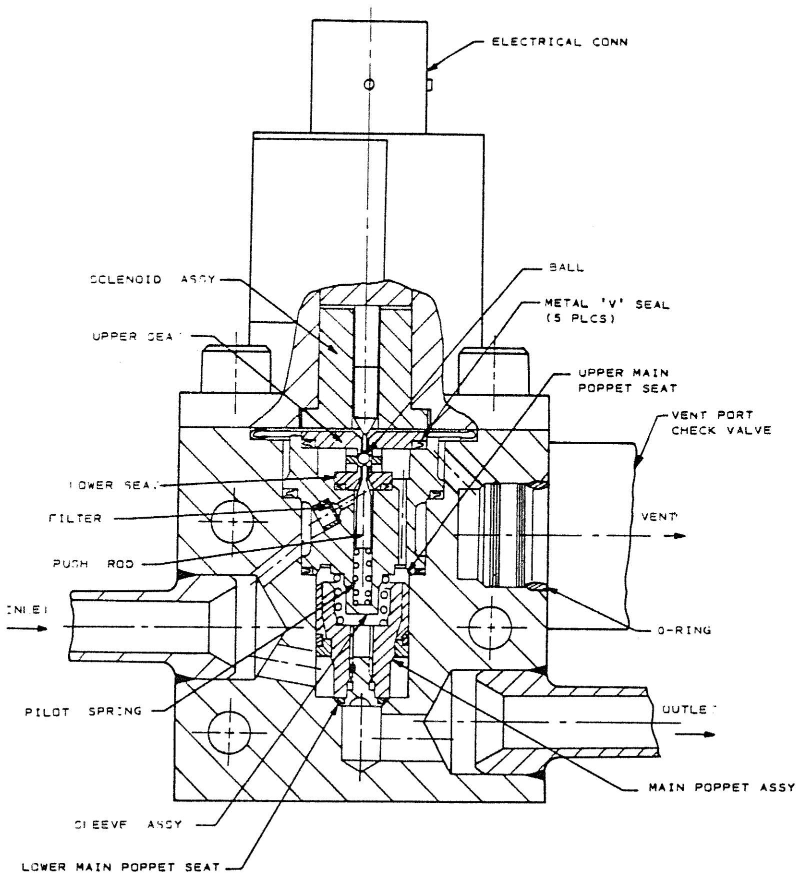
MC284-0403-0002 3/8 IN 850 PSI
MC284-0403-0003 1/2 IN 850 PSI

DASH NO	USE
0002	1. LH2 FEED MANIF RTLS REPRESS VALVE NO 1 & NO 2
	2. GH2 PRESS LINE VENT VALVE
	3. HELIUM SUPPLY BLOWDOWN VALVE NO 1 & NO 2
0003	1. L ENGINE HE CROSSOVER VALVE
	2. LO2 MANIF REPRESS VALVE NO 1 & NO 2
	3. LH2 MANIF REPRESS VALVE NO 1 & NO 2

Figure 3.3-13.- Helium control valves.

3.3.2.14 Helium Supply Valves

A schematic of the helium supply valves (i.e., the pneumatic helium supply valves, the center, left, and right SSME helium supply valves, and the center, left, and right SSME helium inlet and outlet interconnect valves) is given in figure 3.3-14. These normally closed valves are spring-loaded closed when the control solenoid is deenergized and the pressure is equal on both sides of the main poppet. When the control solenoid is energized, the pilot poppet is moved to the vent position, the pressure on the upper side of the main poppet is vented to ambient, and inlet pressure pushes the main poppet open.



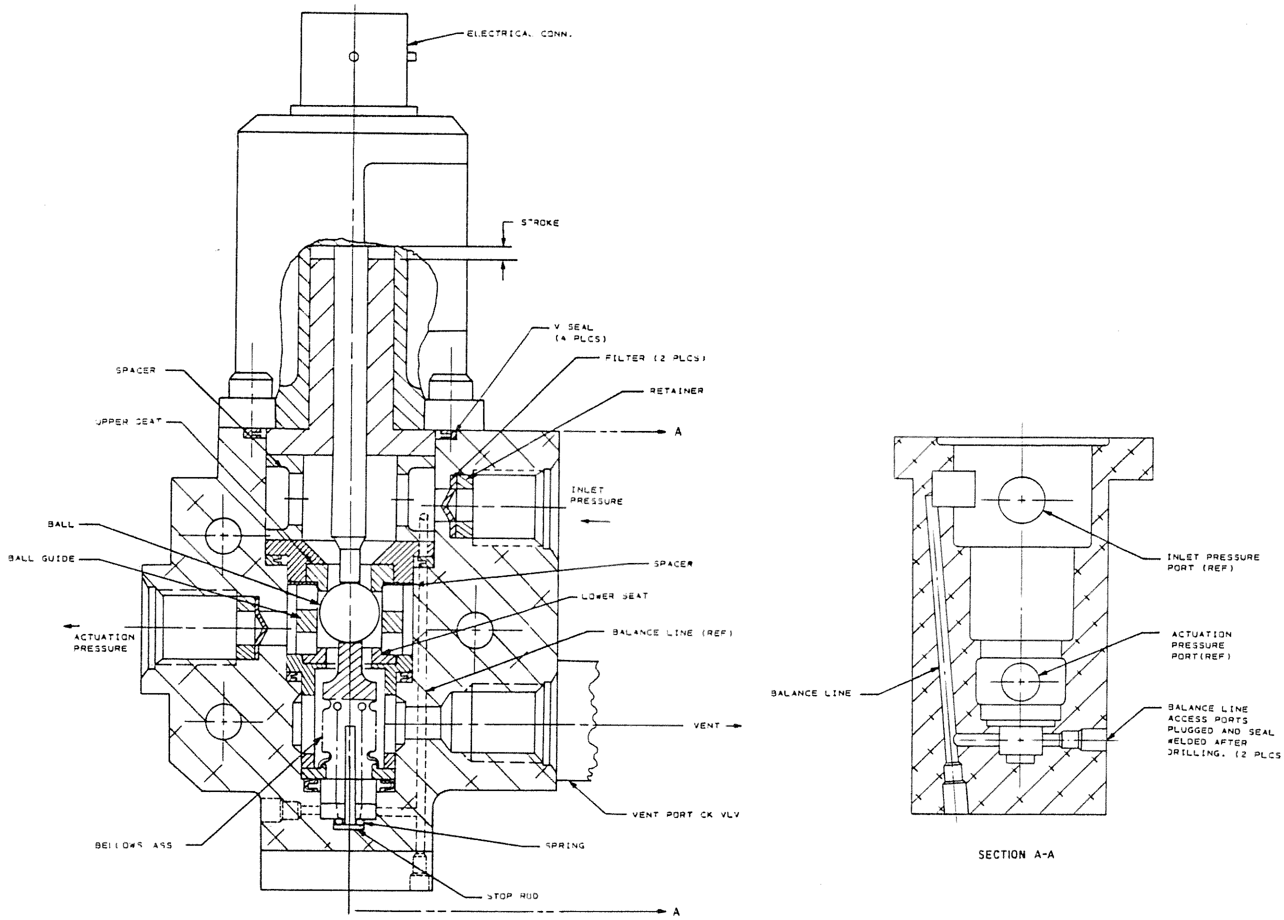
HELIUM SUPPLY VALVES
MC284-0403-0001 3/8 IN 4500 PSI
MC284-0403-0007 1/2 IN 4500 PSI

DASH NO	USE
0001	1. PNEU HELIUM SUPPLY ISOL VALVE NO 1 AND 2 2. C.L.R. HE OUTLET INTERCONNECT VALVE
0007	1. C,L,R ENGINE HE SUPPLY ISOL VALVES A AND B 2. C,L,R HE INLET INTERCONNECT VALVES

Figure 3.3-14.- Helium supply valves.

3.3.2.15 Control Valves For Pneumatic-operated Valves

A schematic of the solenoid control valves used for the various pneumatic-operated valves (i.e., prevalves, fill and drain valves, feedline disconnect and relief isolation valves, LO2 POGO valves, LH2 RTLS dump valves, LO2 OVBD bleed valve, LH2 recirculation valves, LH2 topping valve, and high point bleed valve) appears in figure 3.3-15. These valves are three-way, normally closed, two position, continuous types and are spring-loaded so that in the deenergized position the pressure port is closed and the actuation port is open to the vent port.



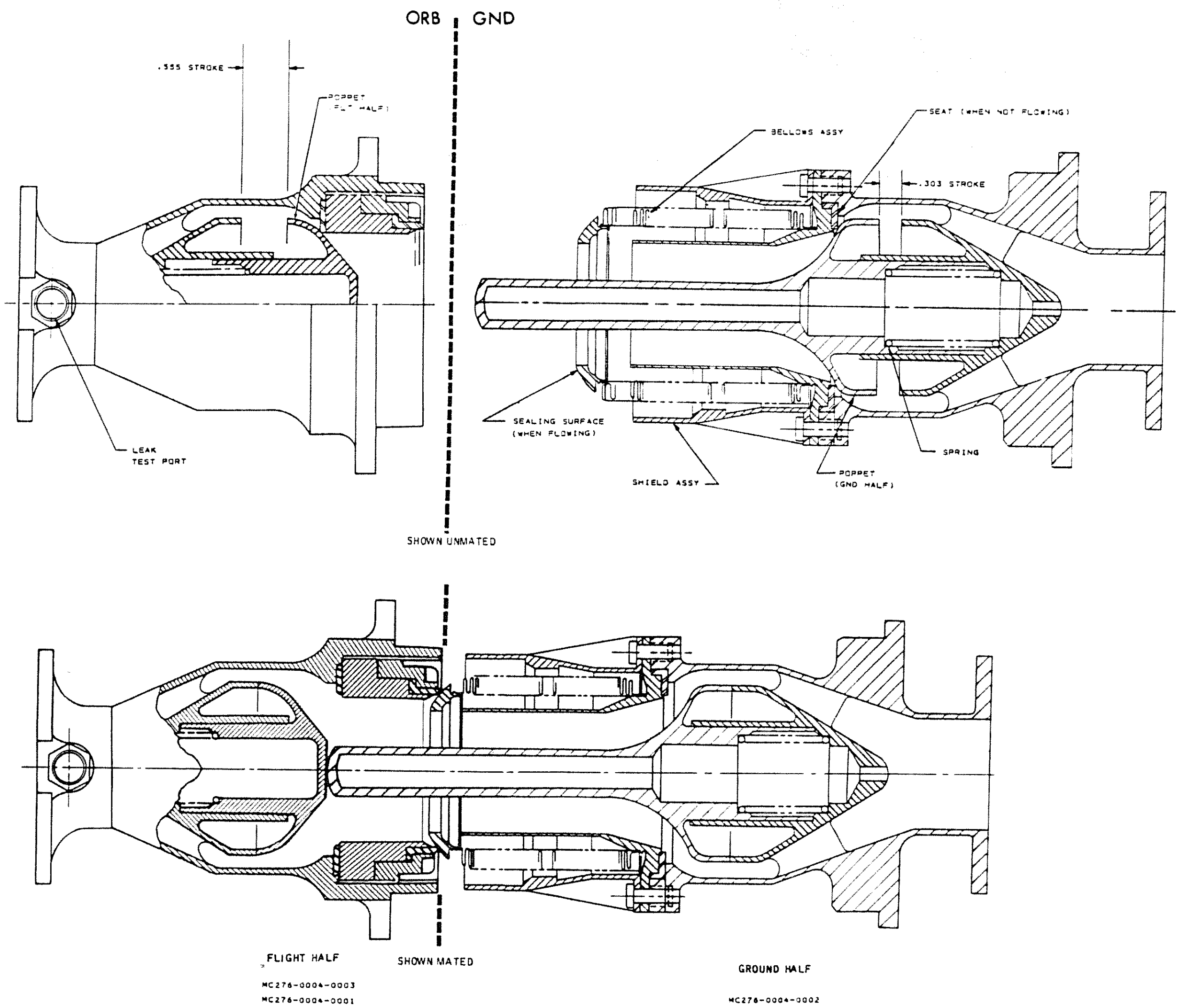
CONTROL VALVES FOR PNEU. OPERATED VALVES
MC284-0404-0001 3/8 IN -850 PSI
MC284-0404-0002 1/4 IN -850 PSI

DASH NO	USE
0001	1. C,L,R ENGINE LO2 PREVALVE OPEN SOL
	2. C,L,R ENGINE LO2 PREVALVE CLOSE SOL
	3. LO2 FEEDLINE DISC VALVE OPEN SOL
	4. LO2 FEEDLINE DISC VALVE CLOSE SOL
	5. C,L,R ENGINE LH2 PREVALVE OPEN SOL
	6. C,L,R ENGINE LH2 PREVALVE CLOSE SOL
	7. LH2 FEEDLINE DISC VALVE OPEN SOL
	8. LH2 FEEDLINE DISC VALVE CLOSE SOL
0002	1. LO2 INBD AND OUTBD FILL AND DRAIN OPEN SOL
	2. LO2 INBD AND OUTBD FILL AND DRAIN CLOSE SOL
	3. LO2 POGO ACCUM RECIRC VALVE NO 1 AND NO 2 CLOSE SOL
	4. LO2 OVBD BLEED VALVE CLOSE SOL
	5. LO2 FEEDLINE RELIEF ISOLATION VALVE CLOSE SOL
	6. LH2 INBD AND OUTBD FILL AND DRAIN OPEN SOL
	7. LH2 INBD AND OUTBD FILL AND DRAIN CLOSE SOL
	8. LH2 RECIRCULATION PUMP VALVES OPEN SOL
	9. LH2 RECIRCULATION DISC VALVE OPEN SOL
	10. LH2 RECIRCULATION DISC VALVE CLOSE SOL
	11. LH2 REPLENISH (TOPPING) VALVE OPEN SOL
	12. LH2 FEEDLINE RELIEF S/OFF VALVE CLOSE SOL
	13. LH2 RTLS INBD AND OUTBD DUMP VALE OPEN SOL
	14. MPS HIGH POINT BLEED VALVE OPEN SOL

Figure 3.3-15.- Control valves for pneumatic operated valves.

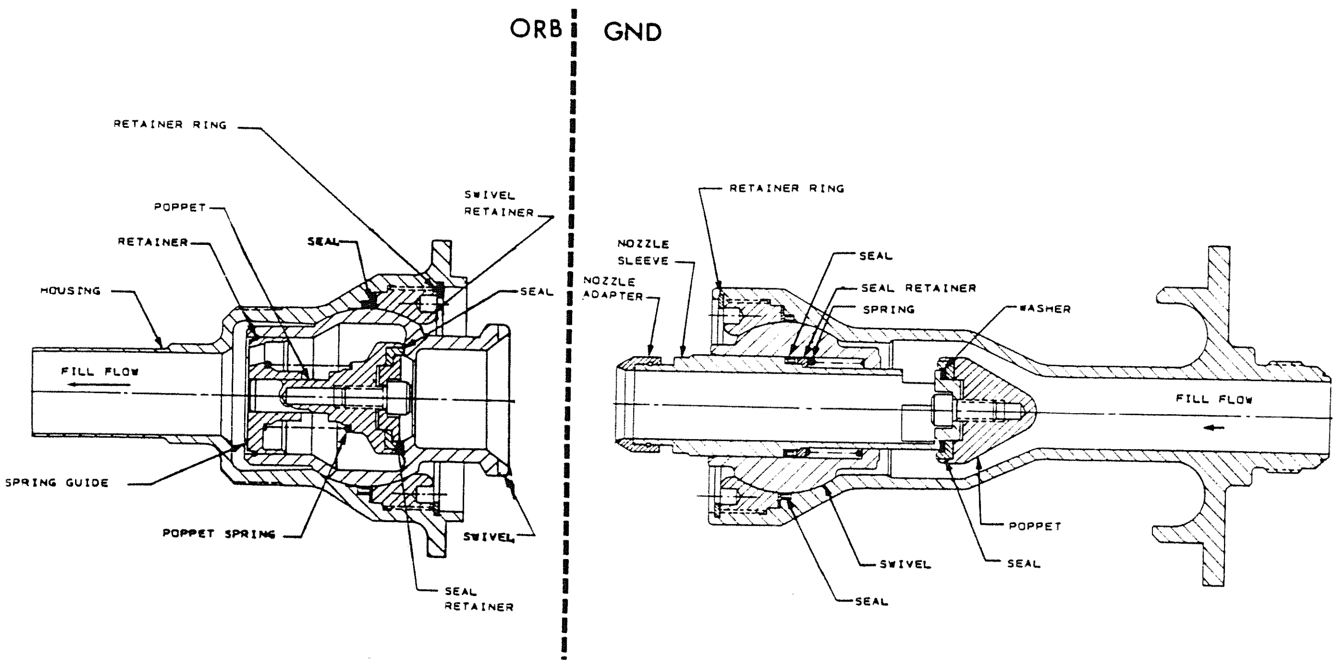
3.3.2.16 Orbiter Ground Disconnects

A schematic of the Orbiter ground disconnects (i.e., LO2 OVBD and high point bleed disconnects, helium disconnects, and fill and drain disconnects) is given in figures 3.3-16 and 3.3-17. The LO2 OVBD and high point bleed disconnects are pushed open during engagement of the Orbiter and ground disconnects. The helium disconnect Orbiter section is actuated open during pressurization by line pressure and the ground section is pushed open by engagement of the two sections. The LO2 OVBD, high point bleed, and helium disconnects are spring-loaded closed when unmated.



DASH NO	USE
0001 AND 0002	1. LO_2 OVBD BLEED DISC
0003 AND 0002	2. MPS HIGH POINT BLEED DISC

Figure 3.3-16.- Overboard and high point bleed disconnect valves.



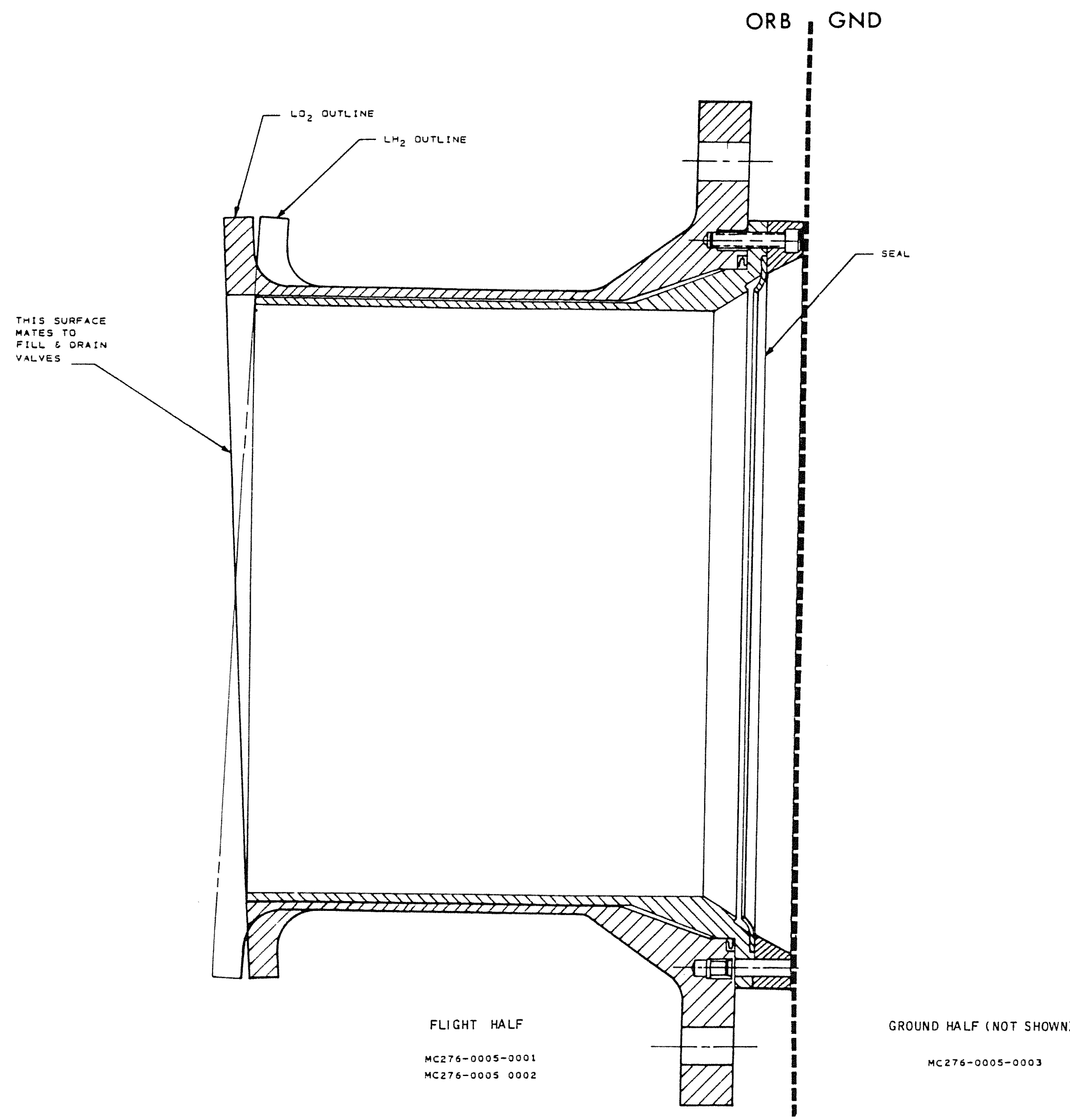
FLIGHT HALF
(SHOWN CLOSED)
MC276-0003-0001

GROUND HALF
(SHOWN OPEN)
MC276-0003-0002

HELIUM DISCONNECTS

DASH NO	USE
0001 AND 0002	1. HELIUM FILL DISC
	2. ET LO_2 TANK PREPRESS DISC
	3. ET LH_2 TANK PREPRESS DISC

Figure 3.3-17.- Helium disconnect valves.



FILL AND DRAIN DISCONNECTS

DASH NO	USE
0001	1. LO_2 FILL AND DRAIN DISC
0002	1. LH_2 FILL AND DRAIN DISC
0003	1. GND HALF LO_2 AND LH_2 DISC

Figure 3.3-18.- Fill and drain disconnect valves.

3.4 MPS LOW LO_2 AND LH_2 NPSP

3.4.1. General

Liquid oxygen and hydrogen NPSP represents the difference between the head pressure at the pump inlet and the liquid vapor pressure, which is a function of inlet temperature. The low pressure pumps and the high pressure pumps become more susceptible to cavitation as the inlet pressure approaches the vapor pressure. The NPSP requirements are designed to allow sufficient margin between the inlet pressure and the vapor pressure to keep the fluid pressure above the vapor pressure after work is done on the liquid by the low pressure and high pressure pumps. Adequate NPSP prevents pump cavitation and the resulting loss in pump efficiency. Pump cavitation also removes the fluid load from the turbopumps. A reduction in the fluid load on a pump can cause the turbine driving that pump to overspeed and come apart, resulting in uncontained engine damage.

The engine inlet pressure is a sum of the external tank ullage pressure plus the fluid head pressure due to the vehicle acceleration plus the fluid dynamic pressure due to the flow of the fluid minus the pipe friction losses. A change in any of these components will change the inlet pressure and the resulting NPSP margin. The NPSP requirement for a turbopump, however, is primarily a function of the engine power level. A discussion follows concerning the effect these components have on the actual NPSP and the engine requirements. The LH_2 and LO_2 systems is dealt with separately because the respective pumps react differently to changes in ullage pressure and vehicle acceleration.

3.4.2 LO_2 NPSP

NPSP for the LO_2 turbopumps is critical at MECO. This NPSP criticality primarily is because of the effects that acceleration has on head pressure at the engine interface. The acceleration on the high density LO_2 can create as much as 130 psi difference in head pressure. The 20 to 22 psia ullage pressure has only a small effect because the vapor pressure of the LO_2 (16 psia) will make up a significant part of the ullage pressure. The three-control-valve pressurization configuration has been replaced with a fixed-orifice configuration. If a tank rupture occurs, a loss of all ullage pressure would reduce the NPSP by 20 psi, but it would still be above the required NPSP of 7.8 and 13.22 psi at 100 and 104 percent power level respectively. Figure 3.4-1 shows a typical NPSP profile as a function of flight time. The curve shows that NPSP typically varies from 30 psi to 145 psi as the vehicle's g level builds from 1g to 3g's for an engine at 100 percent power level. A 20-psi reduction would drop the NPSP to approximately 10 psi, which is still above the 7.8-psi requirement.

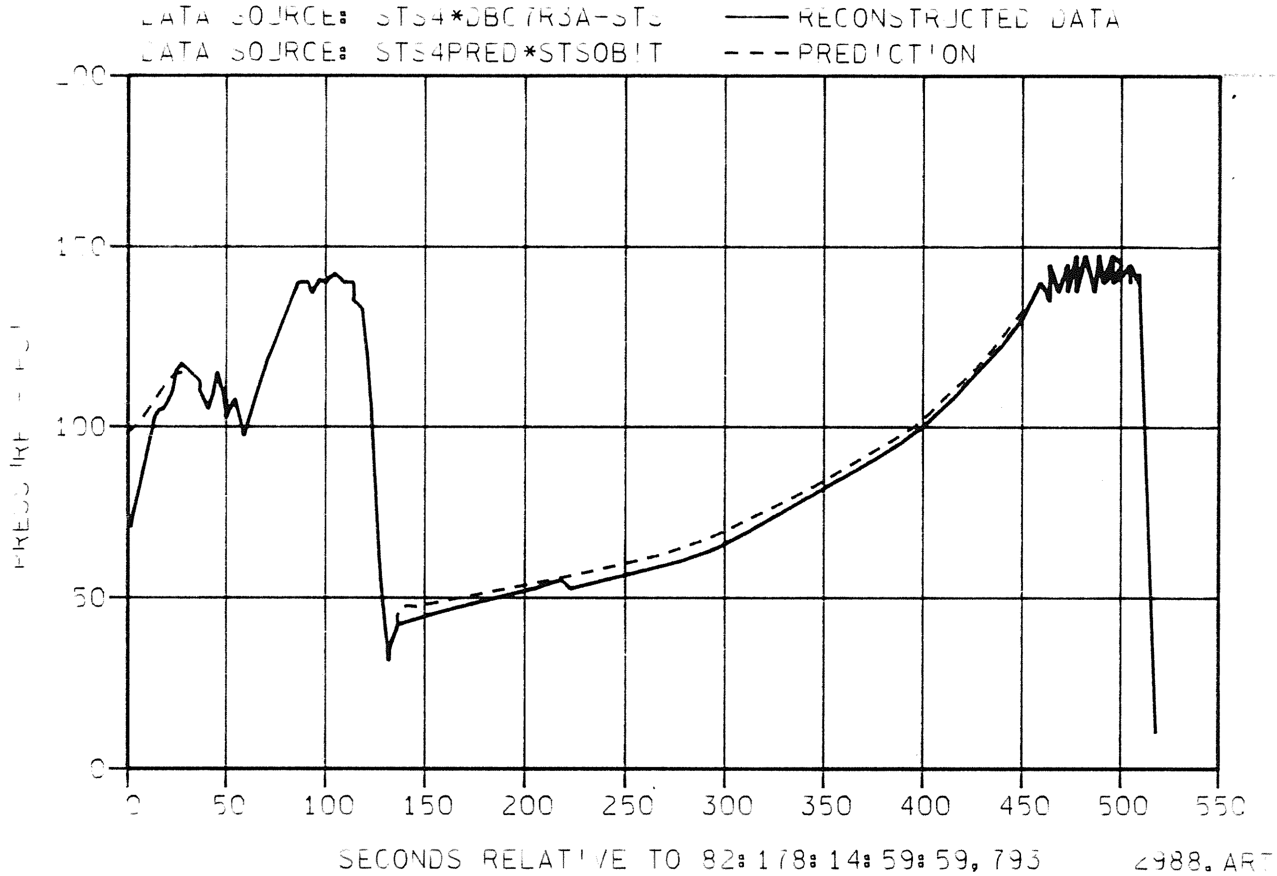


Figure 3.4-1.- LO_2 NPSP at Orbiter-SSME no. 2 interface.

3.4.2.1 LO_2 NPSP at SSME Shutdown

There will always be acceleration forces on the LO_2 while at least one engine is operating. At main engine cutoff however, all three engines are shut down at the same time. As thrust decays, the acceleration on the vehicle quickly drops to zero. The NPSP on the low pressure and high pressure LO_2 pumps will also quickly go to zero psia. The Space Shuttle main engines are designed to shut down with a hydrogen-rich environment. The main fuel valve remains open 5 seconds after the main oxidizer valve closes. Fuel flowing into the high pressure oxidizer turbopump (HPOT) preburner will continue to drive the turbine. The driven turbine is coupled to an unloaded pump which could easily cause the turbine to overspeed and fly apart. To prevent overspeeding, an artificial head pressure is applied to the HPOT. Helium is used to pressurize the POGO accumulator between the low and high pressure pumps. The pressurization of this line creates a temporary head on the high pressure pump, which helps to slow the pump while the fuel-rich flow is going through the preburner. Helium pressure is applied in the following manner: after the MECO command flag is set, the MPS pneumatic crossover valve is opened. This valve allows the left helium supply to act as a redundant pneumatic source for the three LO_2 prevalve actuators. The pneumatics tank is the primary supply. The prevalve close time delay of one second is also started when MECO is commanded. After the timer expires, the closed. The pneumatics tank is also used to supplement the engine helium by opening the three in-interconnect valves. The in-interconnect valve for each engine will open for 20 seconds unless an engine has prematurely shut down. After 1.75 seconds past the MECO command, the helium precharge valve on each operating engine opens to allow a 4-second pressurization of the pogo accumulator. The avionics and hardware are fail-operational. Two failures are required to prevent the prevalve from closing. Prevalve closure and helium pressurization are not required for an engine which shuts down prematurely. The remaining two engines provide enough vehicle acceleration to ensure a safe shutdown.

3.4.2.2 NPSP Shutdown Testing

Rocketdyne has performed several tests to study engine shutdown characteristics with and without prevalve closure and helium pressurization. To this date no engine has been tested with zero psi NPSP at shutdown. One uncontrolled test was run at 5 psi NPSP with no adverse effects, but because of the uncontrolled nature of the test, the results were not conclusive. All analytical data indicate that LO_2 prevalves need not close if the NPSP is above 20 psi before shutdown. Under normal MECO conditions, the NPSP is well above 20 psi. This requirement could be violated if an LO_2 low level shutdown should occur.

3.4.2.3 LO_2 Low Level Shutdown

The SSME operations sequence is designed to shut down the three engines after an appropriate time delay when the liquid/gas interface is detected. The interface is sensed when two of four LO_2 low level sensors in the orbiter feedline go dry. At this time, one of three timers is started. The timer to be used is determined by the abort mode selected and the number of engines operating. When the selected timer runs out, MECO is automatically commanded. A low level MECO will occur if an ascent performance problem exists that causes LO_2 to be depleted before a guided MECO. Problems such as an engine out near press-to-MECO, or an engine problem that could cause a high mixture ratio shift, or low SRB performance during first stage, could cause an LO_2 low level MECO. Other factors such as large wind loads can contribute to a low level shutdown.

The three timers are designed to maximize the amount of LO_2 burned in the engines without violating the NPSP requirements at the operating power level and to minimize the resulting underspeed at MECO. For example, the STS-5 RTLS timer was set at 0 seconds and is designed for one engine out and two engines throttled down to 67 percent before MECO. The press-to-MECO timer is also set at 0 second but it is designed for two engines operating at 91 percent power level at MECO. The nominal timer is set at 0.478 second and is designed for three engines operating at 65 percent at MECO.

The timers are designed to account for the higher total flow rate occurring when three engines are running instead of two. The largest influence, how ever, is the higher NPSP required for higher power levels. Figure 3.4-2 shows how the NPSP requirement increases as the power level increases. The timers are designed to meet this NPSP requirement. These requirements ensure against gas entering the engine, possibly causing pump cavitation and overspeeding.

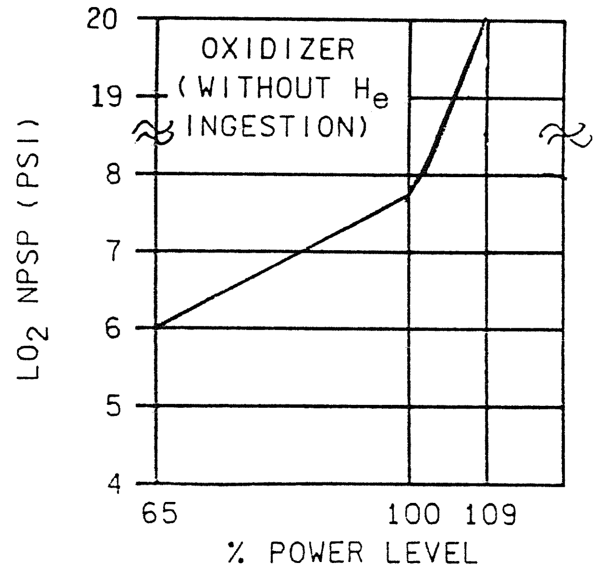


Figure 3.4-2.- Minimum steady state LO_2 NPSP.

The nominal three-engine timer is more critical than the other two. On performance-critical missions, which fly heavy payloads or high orbit inclinations, any performance problem could cause a low level MECO. The propellant margins at MECO become very critical on the three-engine case. A violation of the LO_2 NPSP could occur at MECO if one of the three engines had previously experienced a throttling problem such as a hydraulic or electrical lockup. If one engine's throttle sticks at a high power level, while the other two engines throttle down, the liquid/gas interface associated with a low-level MECO could be drawn into the engines before MECO is commanded. To prevent this from happening, the flight dynamics officer (FIDO) will evaluate the delta velocity margins after SRB separation as an indication of vehicle performance. If vehicle performance is low and one engine's throttles are stuck at a high level (any level above 67 percent), the booster systems engineer (Booster) will inform the flight director that the crew needs to shut down the stuck engine when the relative velocity reaches 23,000 fps, which is 20 seconds before MECO. If a stuck engine occurs after the 23K fps velocity, the crew will not be able to react. A Level II decision relieves the booster engineer of responsibility after 23K fps.

The crew shuts down the stuck engine with the ac power switches and pushes the shutdown pushbutton to mode guidance to two engine control. After guidance is reconfigured, the two remaining engines will be throttled back up to the mission operating level if vehicle acceleration is below 3 g's. The low level timer will be changed to correspond to two engines operating at the mission power level. The power level and engine configuration will then correspond to the timer so that the NPSP at MECO will not be violated. This procedure is only required for the three-engine case, since RTLS performance is not normally a problem and the two engine press-to-MECO timer is the resulting case when the stuck engine is shut down on the three-engine case.

The engine cutoff sensors (ECO) are located on the orbiter and, therefore, save the expense of losing the sensors every time an external tank is separated from the Orbitor. The flow volume below the sensors is small and, therefore, restricts the timer setting. The probability of an LO_2 low level shutdown will also be higher because of the effect heavy payloads have on vehicle performance. The manual shutdown procedure will, therefore, become more probable. An automatic engine shutdown has been recommended for the stuck throttle situation. This software will not be implemented until after a zero psi NPSP test is conducted and it is proven that a LO_2 depletion is unsafe. Even though this test has been approved, there is not enough spare hardware to be risked in such testing. Consequently, the depletion test has not been scheduled.

3.4.2.4 LO_2 Net Positive Suction Pressure Management Decisions

The purpose of this section is to document the decisions concerning LO_2 NPSP that were made in the summer of 1988.

During an audit of the section 5 flight rules it was discovered that flight rule 5-38, MANUAL THROTTLEDOWN FOR LO_2 NPSP PROTECTION AT SHUTDOWN, may not be sufficient to cover all possible low NPSP cases at MECO.

This issue was discussed at the Ascent Flight Techniques Panel on 4/22/88 and 5/21/88, and subsequently discussed at the 7/19/88 PRCB. However, no documentation of these discussions is available. Therefore, Booster personnel attempted to reconstruct the thought processes and final decision. Slightly different NPSP values were provided on 6/11/92 for the fixed-orifice pressurization configuration currently used.

Flight rule 5-38 says that for two engines that prematurely shut down, the remaining engine will be throttled down to 67 percent power level at 1 percent propellant remaining. The purpose of this rule is to reduce the power level which reduces the required NPSP for a safe engine shutdown. Figure 3.4-3 shows the predicted LO_2 NPSP at shutdown versus the required LO_2 NPSP at shutdown for various throttle settings and number of engines running at MECO. The first column contains the throttle settings for each of the three engines at MECO. The center column contains the predicted NPSP and the required NPSP in brackets. The third column (is for information only) contains the LO_2 residuals at MECO and the required residuals in brackets.

Notice that flight rule 5-38 only covers the case 0,0,67. Here there is a negative NPSP margin of 1.87 NPSP (4.23 - 6.1). In this instance, the required and the actual NPSP are fairly close and the community is confident that this case is safe. Refer to figure 3.4-3.

The real concern about rule 5-38 is that it does not cover the cases 91,0,104 and 104,0,104. These cases are just as likely as the two engine-out cases (all are two failures deep). In the process of investigating these two cases it was discovered that Rocketdyne has demonstrated safe shutdown NPSP's as much as 3 psi below the requirement. During SSME test 336, 2 safe shutdown occurred from 100 percent power level (rated power level - RPL) with an NPSP of 5.3 psi. The requirement for shutdown is approximately 8 psi. This demonstrates that a negative 3 margin is probably safe. Although Rocketdyne has <u>not accepted this on the record</u>, discussions with K. Kan/RKDN revealed that they think that a negative 5 margin is probably safe and that these two cases are probably all right. Rocketdyne feels that the LO_2 NPSP at MECO must be well below the requirement for severe pump cavitation to occur.

Since the LO_2 NPSP will be less than required for the 91,0,104 and 104,0,104 cases, it is highly desirable that the SSME shutdown limits be enabled at MECO. Fortunately, current booster procedures require that the limits be enabled at the earliest possible time to support safe engine operation and MECO. There are two exceptions to this procedure, one of the remaining engines has a data path failure and BFS engage.

In light of the above discussion, flight rule 5-38 is fine as written and no changes are required.

The resultant PRCB decision was that the program will accept the risk of violating the required NPSP of two engines shutting down at MECO rather than shutting down a second engine and risk single engine roll control.

3.4.3 <u>LH_2 NPSP</u>

Unlike the LO_2 feed system, the LH_2 propellant system requires critical control of ullage tank pressure and fluid temperature to maintain the required NPSP. A rise in LH_2 temperature could drop fluid density to a value where vaporization in a pump could occur. A sprayed-on insulation is applied to both the LO_2 and LH_2 tanks to reduce aerodynamic and engine plume heating of the fluids. Figure 3.4-3 shows the maximum interface control document (ICD) limit on the LH_2.

REASSESS MPS LO_2 LOW LEVEL CUTOFF (LLCO) PERFORMANCE

- Issue reassess LLCO with SSME'S S/D @ higher than design power levels
  - Design power levels: 3 SSME MECO, all modes - 67%PL
    2 SSME MECO, RTLS - 67%PL / TAL & PTM - 91%PL
  - Fixed-orifice pressurization

- Discussion
  - MECO NPSP & POST S/D residual were provided on 6/11/92
  - RI/MPS reevaluates 51F LLCO reconstructed data
    - O Subcooled LO_2 improves NPSP margin
      - O Increasing predicted NPSP by 1.9 PSI for all cases
  - updated NPSP'S (terminal drain test data + 1.9 PSI):

< >: Req'ts

MECO PL'S(%)	NPSP(PSI)	residual(lbm)	
0,0,67	4.23 <6.1>	3119	<1160>
0,0,91	4.03 <7.33>	2737	
0,0,104	3.19 <13.22>	2535	
0,0,109	3.03 <20>	2456	
91,0,91	√8.9 <7.33>	1446	<700>
104,0,104	8.2 <13.22>	902	Propellant depletion during S/D
109,0,109	6.7 <20>	536	
67,0,104	6.9 <13.22>	1635	
67,0,109	6.4 <20>	1550	
91,0,104	7.4 <13.22>	1097	
91,0,109	7.9 <20>	1201	
67,67,67	√8.65 <6.1>	1232	<240>
67,67,72	√7.95 <6.36>	1112	<240>

Figure 3.4-3.- LO2 NPSP at shutdown.

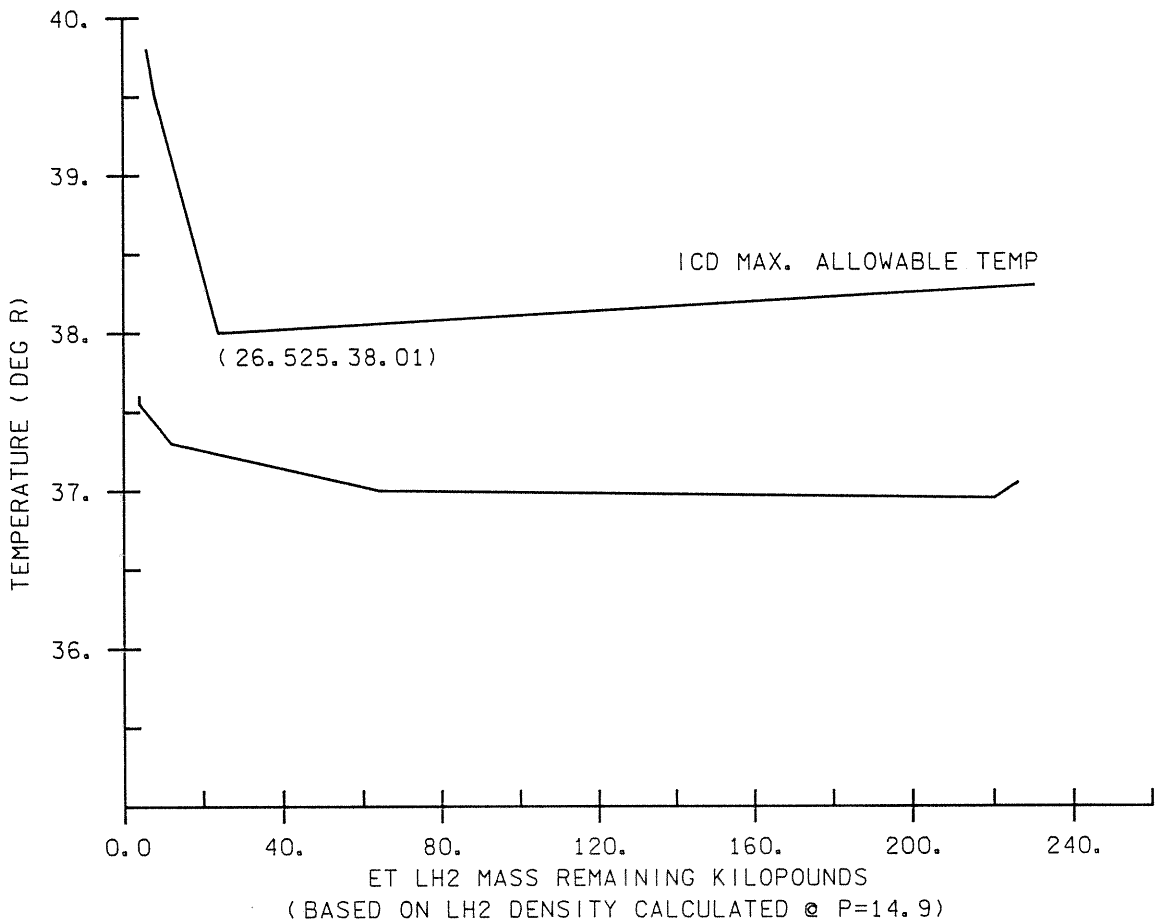
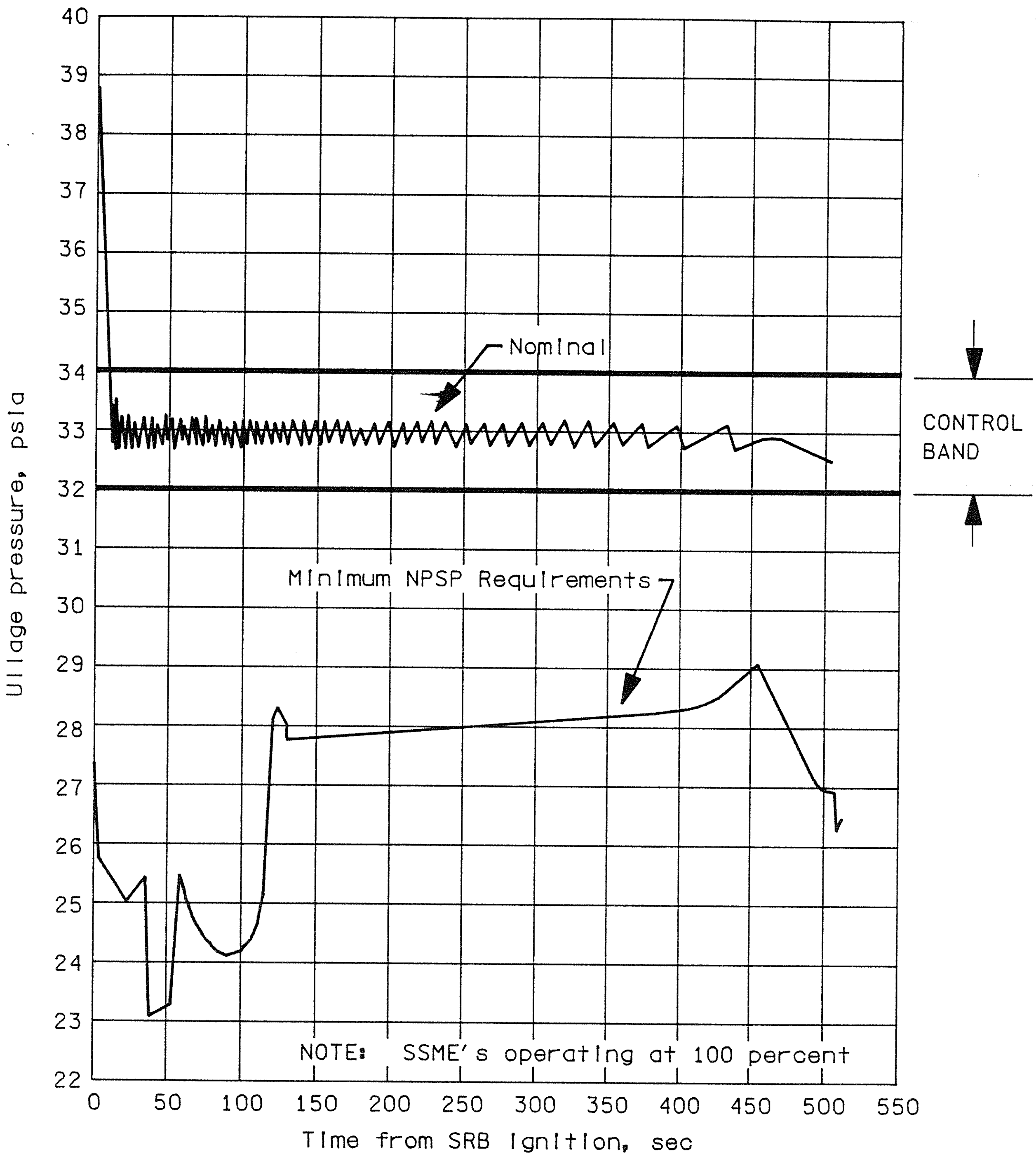


Figure 3.4-4.- ET LH_2 propellant condition during run.

Vehicle acceleration and fluid dynamics have only a small effect on the LH_2 because of its low density. Dynamic effects only contribute about 2 psi to total head pressure. Ullage pressure is maintained between 32 to 34 psia by three flow control valves. This control band can be maintained with only two of the three flow control valves working. Loss of two control valves however, would reduce the pressurization flow so that the ullage pressure profile could not be maintained. Figures 3.4-5, 3.4-6, and 3.4-7 show the predicted ullage profile for the lightweight tank supplying three SSME's at 100 percent, 104-percent, and 109-percent power levels. Also shown is the minimum ullage pressure needed to maintain minimum LH_2 NPSP requirements.

Figure 3.4-8 illustrates how the ullage pressure profile would decay if two flow control valves failed closed at lift-off. At approximately 220 seconds, ullage pressure would decay below 28 psi, too low to maintain the 5.4 psi NPSP at 104 percent power level. Flow valves would remain closed if the ullage pressure transducer failed above 34 psi or if the valve stuck in the closed position.

The orbiter crew has a class three alert and fault summary message driven by the backup flight system (BFS) guidance and annunciation software that is triggered any time an ullage pressure transducer drops below 31.6 psia. The crew can look at the BFS Sys Summ 1 display to check ullage pressures. If two sensors read above 34 psia and the third is less than 31.6 psia, the crew should suspect that two ullage pressure sensors have failed high and the two corresponding flow control valves are closed. After SRB separation the crew will throw the main propulsion system LH_2 ULL PRESS switch to open. The switch is only thrown open after SRB separation to prevent H_2 venting and burning during first stage. The switch is positioned back to AUTO when the true ullage pressure builds to 34.5 psia.



2930.ART;2

Figure 3.4-5.- Predicted LH_2 tank ullage pressure-LWT.

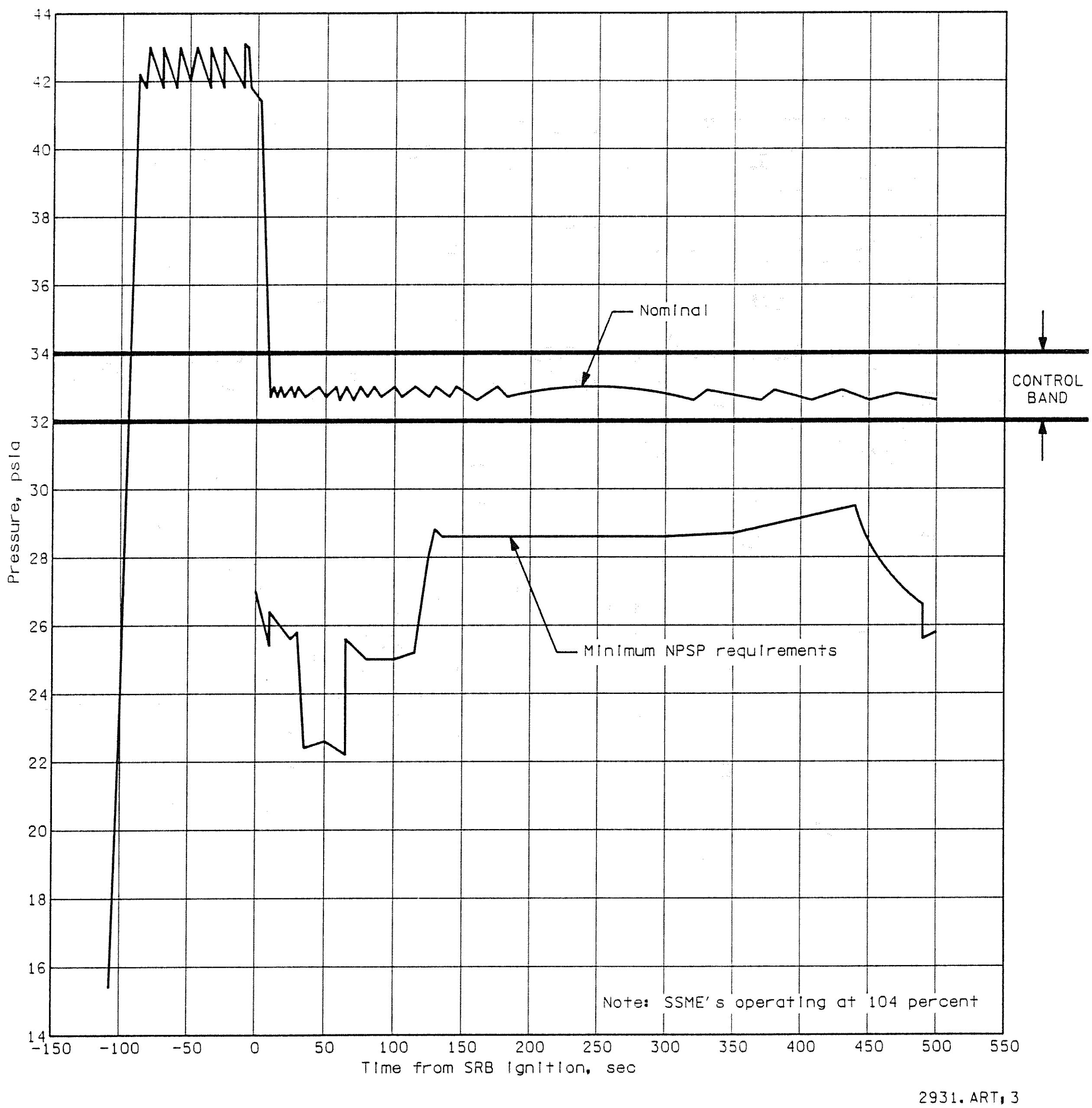


Figure 3.4-6.- Normal liquid hydrogen pressurization performance.

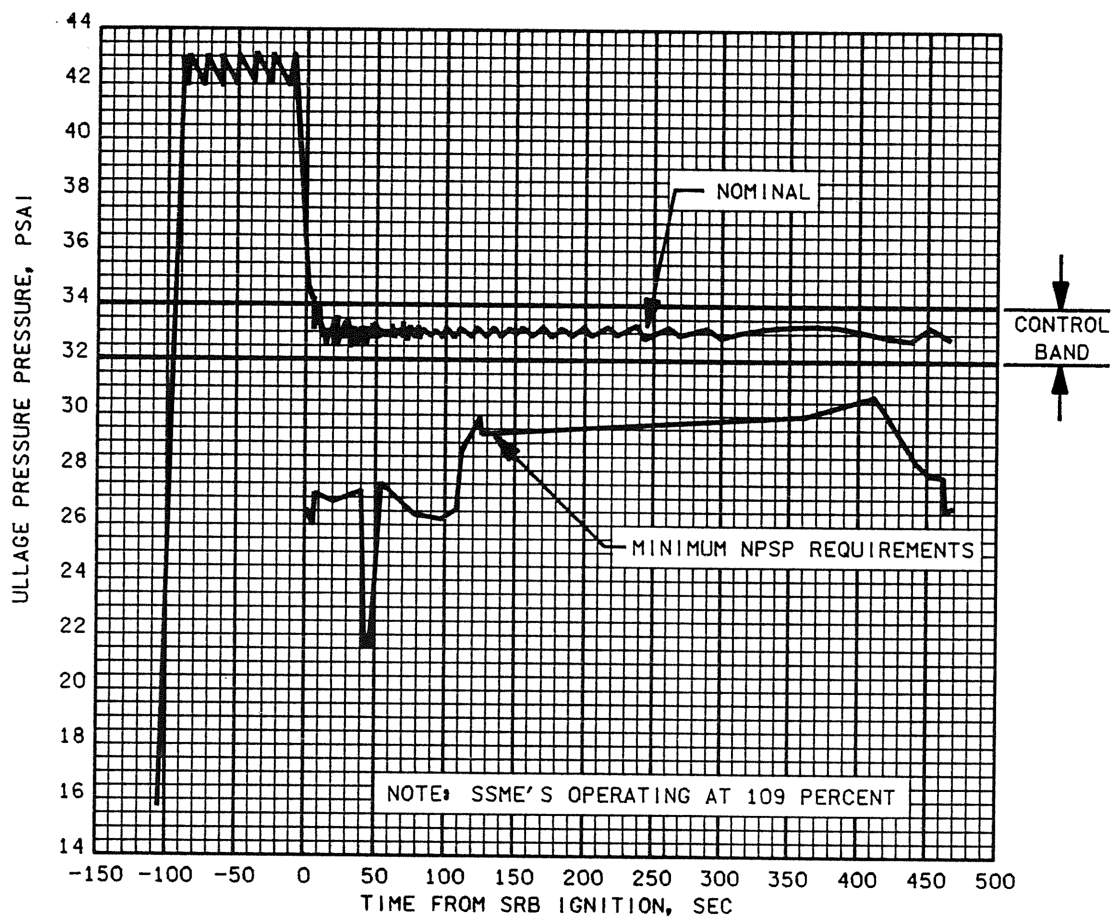


Figure 3.4-7.- Normal liquid hydrogen pressurization performance.with light weight tank.

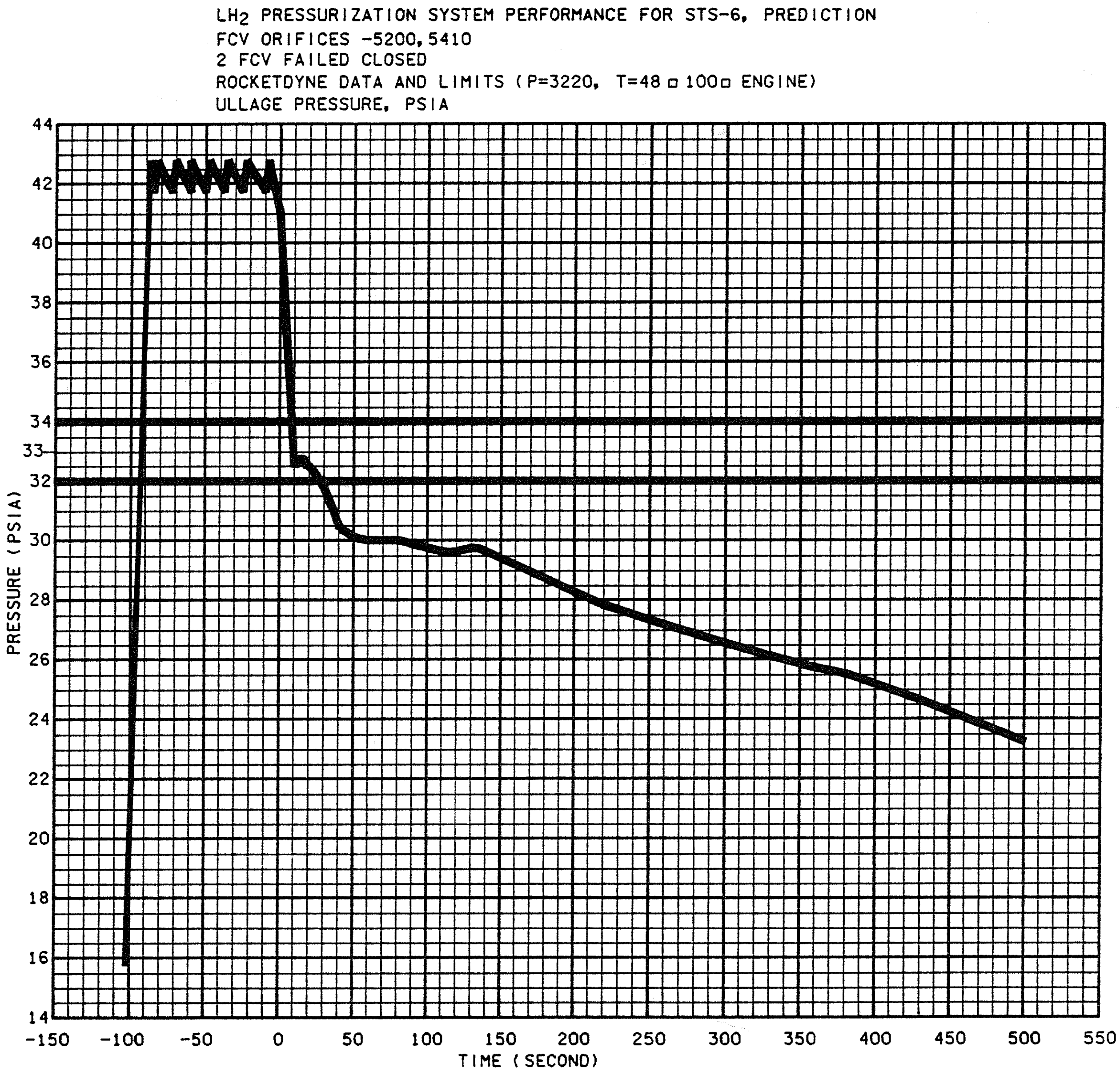


Figure 3.4-8.- LH_2 tank ullage pressure, STS-6
two FCV failed closed.

A direct failure of two flow control valves to open causes all three ullage pressure transducers to read low. If three of three transducers are below 31.6 psia, the crew will throw the ullage pressure switch to open. This overrides the automatic mode and applies open power to all three valves. If the three pressures continue to fall, the Mission Control Center (MCC) will ask the crew to throttle down to 65 percent power level, which will reduce the friction drop in the flow lines and bring the NPSP up. Also, the required NPSP at 65 percent is reduced to 4.8 psi. Figure 3.4-8 shows the minimum LH_2 NPSP as a function of power level. No other action can be taken after throttle down. The possibility of losing three SSME's exists if the ullage pressure continues to drop and the NPSP requirement is violated.

Engine shutdown of at least one engine would probably be automatic after the High Pressure Fuel Turbopump (HPFT) or HPOT turbine discharge temperatures violated their shutdown limits. The high turbine temperatures would occur because of a reduction in H_2 flow when the HPFT lost efficiency. If the other engines did not safely shut down, their automatic shutdown limits would be inhibited when the first engine went down. A subsequent violation of the NPSP requirements to these engines could result in uncontained engine damage if their shutdown limits remained inhibited.

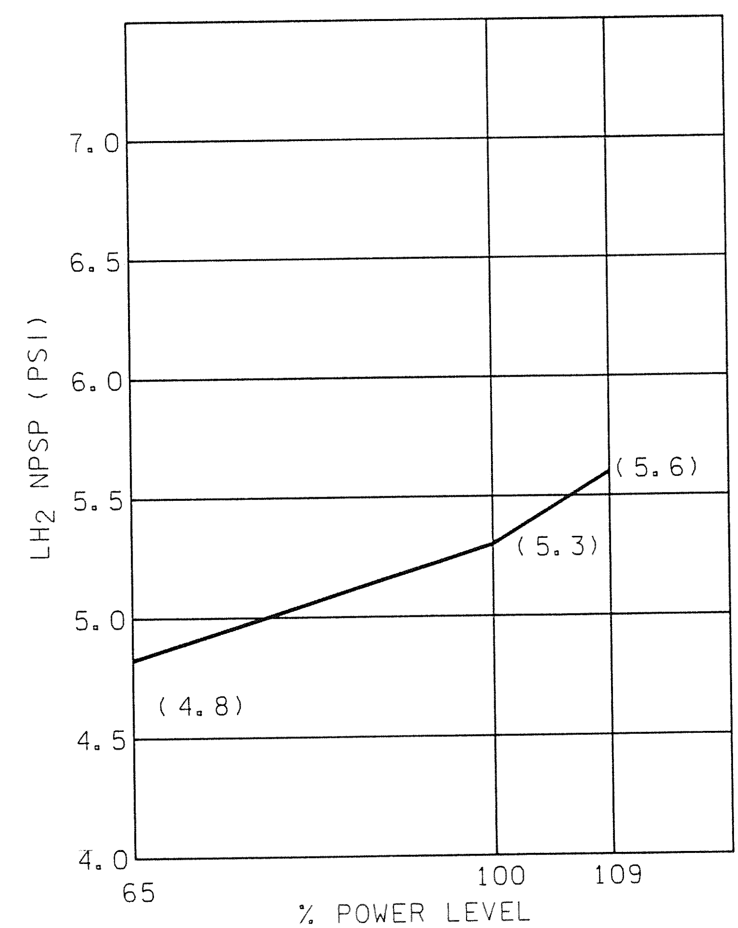


Figure 3.4-9.- Minimum steady state LH_2 NPSP.

3.5 MPS PROPELLANT DUMP

3.5.1 General

The MPS propellant dump is performed following MECO and ET separation to expel the unused LO_2 and LH_2 remaining in the orbiter and SSME LO_2 and LH_2 propellant feedlines. The dump is required to prevent a shift of the orbiter center of gravity (c.g.) in the aft direction that is produced by the location and weight of residual propellants. Non-nominal c.g. locations can adversely affect vehicle control during reentry or RTLS abort maneuvers. Trapped LH_2 remaining in the propellant lines gives rise to a hazardous condition, in that it may combine with atmospheric oxygen during reentry and create a potentially explosive mixture. In addition, the propellant dump is required to preclude spurious venting induced by the trapped propellants while in orbit. This venting will produce navigation errors and also could be a potential contamination source for scientific experiments located in the payload bay.

There are three basic dump modes corresponding to the three major mission software modes in which the MPS propellant dump is performed. Propellant dump is performed in MM 104 during a normal, ATO, or AOA mission. Propellant dump during a TAL abort is performed in MM 304, and in MM 602 for RTLS abort. The quantity of LH_2 remaining in the orbiter lines and SSME's is approximately 304 lbm. The maximum LO_2 propellant mass remaining is approximately 3870 lbm and can be as low as 1847 lbm if SSME shutdown is initiated by the LO_2 low level cutoff sensors. The propellant dump paths include the SSME and fill/drain system for the LO_2 propellant, while the LH_2 dump paths are the RTLS dump line, fill/drain, and recirculation/replenish line.

The crew has the capability to manually initiate the dump after MECO confirmed plus 20 seconds. To manually start the dump, the MPS propellant dump sequence switch on panel R2 is placed in the start position. After this manual input, the GPC initiates the LO_2 and LH_2 dump.

Manual dump capability is available in the PASS for the nominal ascent and for all aborts. When the BFS is engaged, a manual dump can only be performed for a TAL abort. A dump that is started manually is stopped automatically at the proper time for the particular ascent, unless the MPS propellant dump sequence switch is placed in stop.

3.5.2 Functional Description

Dumping of the residual MPS propellants is effected by a series of electromechanical devices, mostly from the MPS subsystem, controlled by the GPC's with optional inputs from the crew as a backup. The timeline for the nominal dump is shown in figure 3.5-1.

Operational elements of the MPS subsystem required for the dump consists principally of valves that allow for the expulsion of the residual propellants from the orbiter. Figures 3.5-2 and 3.5-3 for oxidizer and figures 3.5-4, 3.5-5, and 3.5-6 for fuel are included to show the oxidizer and fuel flow during nominal dumping operations. The fuel and LO_2 flow during RTLS dumping is illustrated in figures 3.5-7 and 3.5-8.

The LO_2 and LH_2 dump process for the nominal ascent and the aborts will be described in detail.

3.5.2.1 Nominal MPS Dump

The LO_2 and LH_2 dump sequence is the same for the AOA and ATO as for nominal ascent. Therefore, the dump process outlined for the nominal dump applies to the AOA and ATO aborts.

Following MECO, the LO_2 feedline relief isolation valve is opened and the three SSME main oxidizer valves, the three LO_2 prevalves, and the LO_2 feedline disconnect valve are closed. The residual liquid oxygen that is trapped between these closed valves and which must be dumped is shown in figure 3.5-2. The MPS dump sequence is scheduled when MECO confirmed plus 20 seconds has occurred. The sequence immediately configures the SSME helium supply to provide helium for use in the dump. The left, center, and right helium interconnect out valves are opened and the pneumatic crossover valve is opened.

An LO_2 dump timeline is given in figure 3.5-1. At OMS-1 ignition or crew initiated manual dump start, the LO_2 manifold repress valves 1 and 2 are opened, which provides helium to pressurize the LO_2 manifold to 20 to 25 psig. The LO_2 prevalves and the SSME main oxidizer valves are opened and the LO_2 is expelled through the SSME main combustion chamber. In auto mode 90 seconds after the LO_2 prevalves open, the LO_2 manifold repress valves are closed, and the LO_2 manifold is allowed to blow down for 30 seconds before closing the SSME main oxidizer valves.

Figures 3.5-9 and 3.5-10 present the LO_2 feed system pressure, mass remaining, total flowrate, and total SSME thrust for three SSME dump capability. The prediction is based on STS-4 flight data reconstruction, corrected to reflect the current normal dump sequence. The flight data indicates that the maximum LO_2 dump flowrate at the start of the dump through the three SSME's is approximately 90 lb/sec. This peak flowrate generates a total thrust of 2100 lbf. After the feedline pressure decays to the pressure control band, the steady state value reduces to 35 lb/sec flow and 700 lbf thrust. At approximately 75 seconds into the dump, blowthrough occurs and helium pressurant flows overboard, resulting in loss of feed system pressure control. The LO_2 mass remaining at the end of the dump is estimated to be approximately 200 lb. This LO_2 mass residual, along with the LH_2 mass is vented off to space during subsequent on-orbit vacuum inerting operations.

The LO_2 dump characteristic for a two SSME capability is shown in figures 3.5-11 and 3.5-12 and is based on STS-3 data reconstruction, where an early APU shutdown prohibited dump through one SSME. The dump data is similar to the normal three SSME dump except that the post-dump residual increased from 200 to 900 lb because of the lower flowrate during the dump and trapped propellant in one SSME 12-inch feedline.

The previous LO_2 dump discussions assumed maximum liquid remaining in the feed system. If a low level cutoff occurs where the feed system is only partially full, the duration of the liquid dump will be shortened proportionately, however, the liquid residual will remain approximately the same.

The manual dump mode allows the LO_2 manifold pressurization valves to be open for 120 seconds before closing. Then after an additional 20 seconds, the SSME's main oxidizer valves are closed. The LO_2 dump subsystem flow during the dump is shown in figure 3.5-3. This completes the LO_2 portion of a nominal propellant dump.

The LH_2 dump timeline is provided in figure 3.5-1. The LH_2 feedline relief isolation valve is opened at the same time as the LO_2 feedline relief isolation valve. The three SSME main fuel valves, the three LH_2 prevalves, the recirculation disconnect valve, and the LH_2 feedline disconnect valve are closed. The configuration of the MPS valves and the trapped LH_2 before dump are provided in figure 3.5-4.

At MECO plus 10 seconds, the ET separation sequence opens the two RTLS dump valves for 30 seconds. This allows the rapid pressure rise in the LH_2 manifold to be vented and prevents an overpressurization of the manifold in the event the LH_2 feedline isolation valve failed close. During this 30-second vent, approximately 51 lb of LH_2 is vented overboard. Figures 3.5-13 and 3.15-14 show the pressure history, mass remaining, flowrate, and thrust history during the 30-second vent.

Following MECO, the SSME controllers enter a post-shutdown standby mode and 16 seconds into this new mode, the controllers open the SSME's fuel bleed valve.

OMS-1 ignition or manual dump start causes the LH_2 manifold repress valves 1 and 2 and the LH_2 inboard and outboard fill and drain valves to be opened. The LH_2 will be dumped overboard, as shown in figure 3.5-5, through the inboard and outboard fill/drain valves and out the 8-inch diameter fill/drain line. The inboard fill/drain valve remains open only 6 seconds and then closes to prevent too rapid a venting of LH_2, which could freeze in the overboard line and prevent further LH_2 dumping. During this 6-second transient, approximately 160 lb of LH_2 is dumped overboard. The peak flowrate and thrust generated by the high dump flowrate is approximately 57 lb/sec and 1150 lbf, respectively. At the closing of the inboard fill/drain valve, the LH_2 topping valve and the LH_2 prevalves are opened. The LH_2 now flows from the pressurized LH_2 manifold through the three open prevalves to the SSME's. In each SSME the LH_2 passes through the low and high fuel pressure turbopumps out the fuel bleed valves and back to the orbiter 4-inch LH_2 recirculation line. The open LH_2 topping valve allows the LH_2 to flow to the fill/drain line out the outboard fill/drain valve and overboard as shown in figure 3.5-6. This pressurized dump continues until OMS-1 ignition plus 88 seconds, at which time the LH_2 manifold repress valves 1 and 2 are closed, thereby preventing further helium pressurization of the feedline manifold. The LH_2 system is allowed to blow down for 32 seconds before closing the outboard fill/drain valve, the topping valve, and the SSME bleed valves. Figures 3.5-15 and 3.5-16 present the LH_2 dump performance characteristics, including the feed system pressure and mass remaining history for the normal on-orbit dump. The estimated LH_2 mass remaining in the feed system/SSME is less than 5 lb, which is vented off to space during subsequent on-orbit vacuum inerting operations. The LO_2 and LH_2 dump portion of the sequence is now complete, but the sequence continues by deenergizing MPS valves and configuring the helium system for on-orbit operations. The sequence is descheduled when the transition to OPS 2 occurs.

3.5.2.2 Manual MPS Dump

The objective of the manual dump mode is to provide the crew the means to initiate the dump without an OMS-1 ignition signal. The manual dump mode is required for a delayed OMS-1 burn or a situation requiring dump before the nominal OMS-1 burn.

The MPS dump can be manually initiated after MECO confirmed plus 20 seconds, when the MPS PRPLT dump sequence switch on panel R2 has been set to START. The propellant flow path and the first part of the dump sequence are the same for both the nominal and manual dumps. Figure 3.5-17 shows the sequence for a manual dump and can be used with figure 3.5-1 to compare the differences in the manual and nominal dumps.

3.5.2.3 TAL Propellant Dump (MM 304)

The dump procedure for TAL missions is presented in figure 3.5-18. As in the case of a Normal dump (MM 304), the LH_2 feedline is vented for 73 seconds beginning at MECO plus 10 seconds through the RTLS dump line (figures 3.5-13 and 3.5-14) The dump is software initiated at MM 304 without helium pressurization, as shown in the dump procedure in figure 3.5-18. The LO_2 and LH_2 propellants consequently are dumped under their own internal pressure because of liquid boiling/self pressurization.

The LH_2 dump is initiated by simultaneously opening the RTLS dump valves and inboard and outboard fill and drain valves. The resultant feedline pressure decay, mass remaining, flowrate, and thrust history through the two dump paths are shown in figures 3.5-19 through 3.5-21. The dump characteristics shown in these figures are derived from a reconstruction analysis of a DTO performed on mission 51-D, which simulated the TAL dump operation, and therefore are considered to be accurate. The reconstructed data indicated that the LH_2 dump is completed in 15 seconds, with peak thrust and flowrate values of 1150 lbf and 57 lb/sec generated at the fill/drain line exit, and 65 lbf thrust and 4.6 lb/sec generated at the RTLS dump line exit. Since the LH_2 residual on 51-D DTO could not be measured accurately, the LH_2 mass remaining was estimated to be approximately 3 lbm, using a vacuum inerting two-phase flow model and post-test pressure history and system leakage considerations.

In the event that the LH_2 fill and drain valve fails closed, the dump flowrate capability is greatly reduced. Figures 3.5-22 and 3.5-23 present the dump characteristics for the unpressurized dump through the RTLS dump valve only. The analysis results indicated that within 500 seconds, all of the 304 lb of LH_2 MPS residual can be dumped to the 3 lb LH_2 level, where hydrogen solidification begins. Since the LH_2 dump is terminated automatically at a ground relative velocity (GRV) of 4500 ft/sec, plenty of time exists (>15 minutes) to dump the LH_2 residual in the event of the failure.

The LO_2 MPS propellant is also dumped unpressurized in MM 304. As shown in figure 3.5-18, the LO_2 dump is initiated by the opening of the SSME MOV's. At approximately 320 seconds (GRV 20000ft/sec) the LO_2 fill/drain valves are opened, accelerating the dump. The reason for the delay in opening the fill/drain valves is to relieve the internal pressure through the SSME's and consequently reduce the thrust disturbances generated by the LO_2 fill/drain dump flow on the orbiter. Figures 3.5-24 and 3.5-25 present the LO_2 dump characteristic for a three SSME dump capability. The analysis results indicate that within 175 seconds, all of the LO_2 residual is dumped through the three SSME's. If a two SSME dump capability exists, the duration of the LO_2 dump increases from 175 seconds to 230 seconds. (See figures 3.5-26 and 3.5-27.) In both situations, the LO_2 dump is completed through the SSME's before opening of the fill/drain line, which occurs at MM 304 plus 320 seconds. Consequently, no thrust disturbance forces will be present when the LO_2 fill/drain valves are opened. As a result of dumping all of the LO_2 propellant through the SSME's prior to vacuum inerting through the fill and drain line, fill and drain valve failure does not effect the LO_2 dump residual.

3.5.2.4 RTLS Dump (MM 602)

The dump during an RTLS abort is presented in figure 3.5-28. The RTLS dump is similar to the TAL dump, except for the time between MECO and valve actuations. The expected resulting LH_2 feedline pressure decay, mass remaining, flowrate, and thrust history through the two dump paths are shown in figures 3.5-19 through 3.5-21. As mentioned, STS-51D simulated a TAL dump, and the RTLS dump that was recently changed is expected to look similar to the TAL dump.

In the event that the LH_2 fill/drain valve fails closed, the dump flowrate capability is greatly reduced because the LH_2 residuals can only flow through the 1.5-inch diameter RTLS dump line versus flowing through the 8-inch diameter fill/drain line. Figures 3.5-29 through 3.5-30 present a prediction of pressure, mass remaining, flow, and thrust history for the LH_2 remaining in the system after the inerting operation. Since the RTLS dump is terminated at a GRV of 3800 ft/sec, all the residual LH_2 may not be dumped because of the amount of time available between dump start and dump stop.

The LO_2 dump during the RTLS abort is performed without helium pressurization through the SSME and fill/drain line. In order to minimize the dump thrust disturbances on the side of the orbiter which would result from dump through the fill/drain line, the SSME MOV's are opened first, with the LO_2

fill/drain valves lagging at QBAR >20 psf (MM 602 plus 39 seconds). The SSME MOV's and the LO_2 fill/drain valves are closed simultaneously at GRV 3800 ft/sec by the on-board software. LO_2 dump performance characteristics are presented in figures 3.5-31 through 3.5-33 for three operating SSME's. This analytical prediction has been verified by correlation with data from a special DTO performed on STS-61A, where the RTLS LO_2 dump was simulated. Figure 3.5-34 shows the correlation of measured feed system pressure with the analytical prediction. The results indicate that all of the LO_2 can be dumped within the 150-second time available with peak flowrates of 90 lb/sec through the SSME's and 113 lb/sec through the fill/drain line, and peak thrust levels of 2100 and 1220 lbf at the SSME's and fill/drain, respectively.

In the event that only two SSME's are operational, all of the LO_2 propellant can still be dumped since the LO_2 fill/drain system dump capability is significantly higher than that of the SSME's, and will therefore compensate for dump performance reduction through the SSME's. If, however, the LO_2 fill/drain valve fails to open, some LO_2 residual will remain in the feed system at the end of the dump operation. Figures 3.5-35 and 3.5-36 represent the worst case LO_2 RTLS dump with two SSME's operating plus a failed closed LO_2 fill/drain valve. Approximately 800 lbm of LO_2 remains at the end of the LO_2 dump for the failure case. This LO_2 residual, which is with worst case failures (fill/drain and one SSME), is considered to be acceptable from a vehicle center of gravity, landing weight, and post touchdown venting point of view.

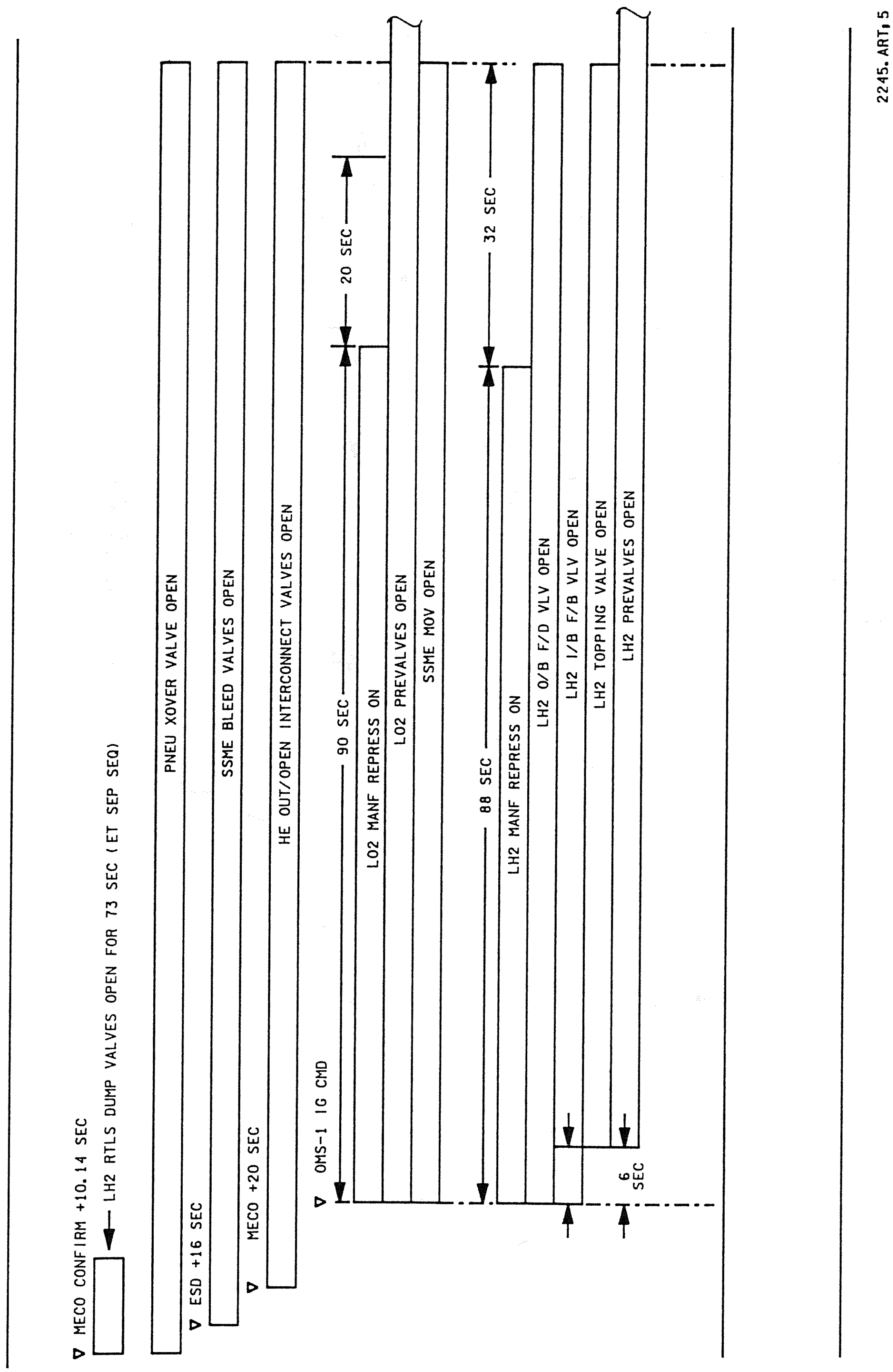


Figure 3.5-1.- Nominal auto MPS dump sequence.

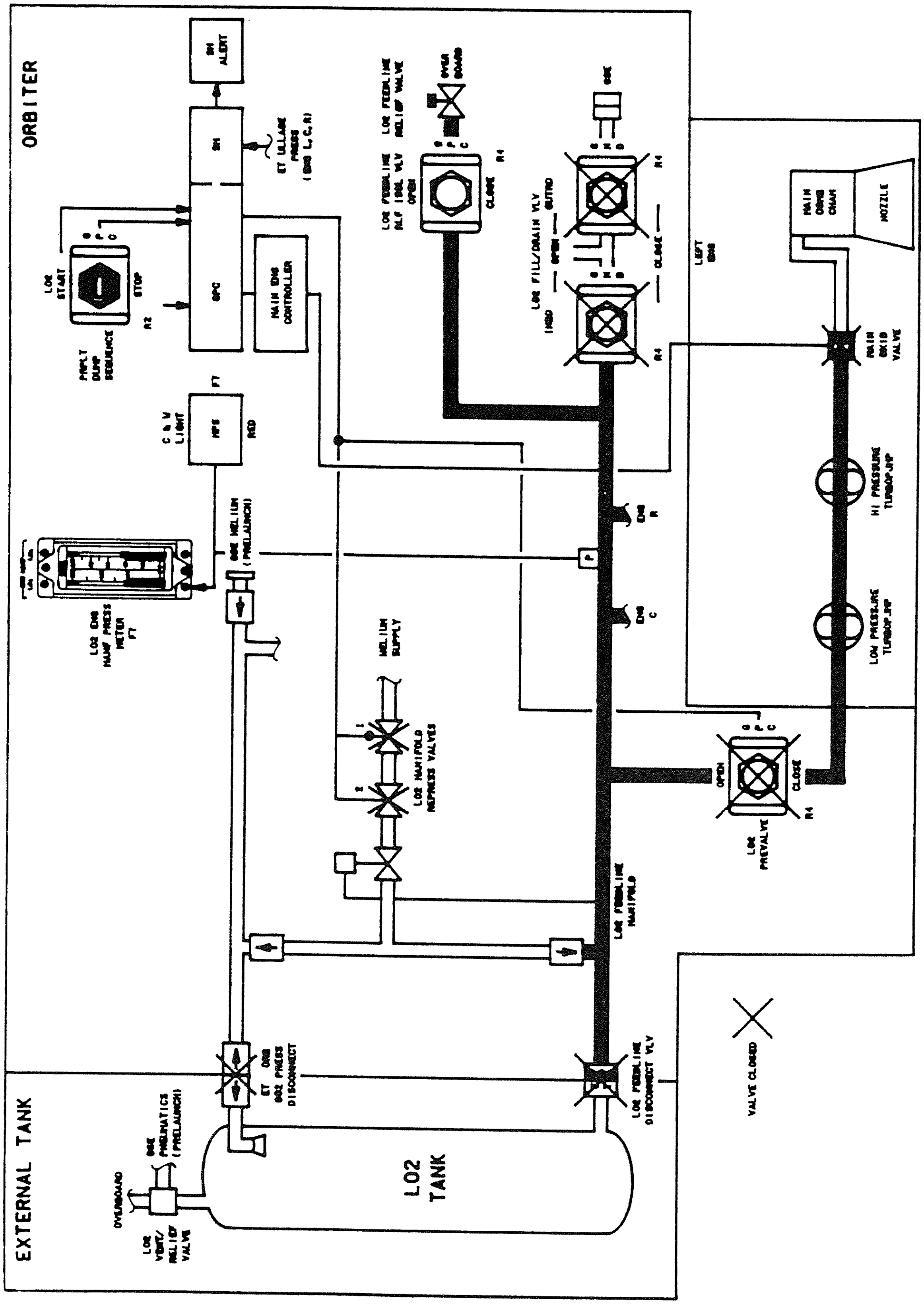


Figure 3.5-2.- MPS LO2 dump subsystem before dump.

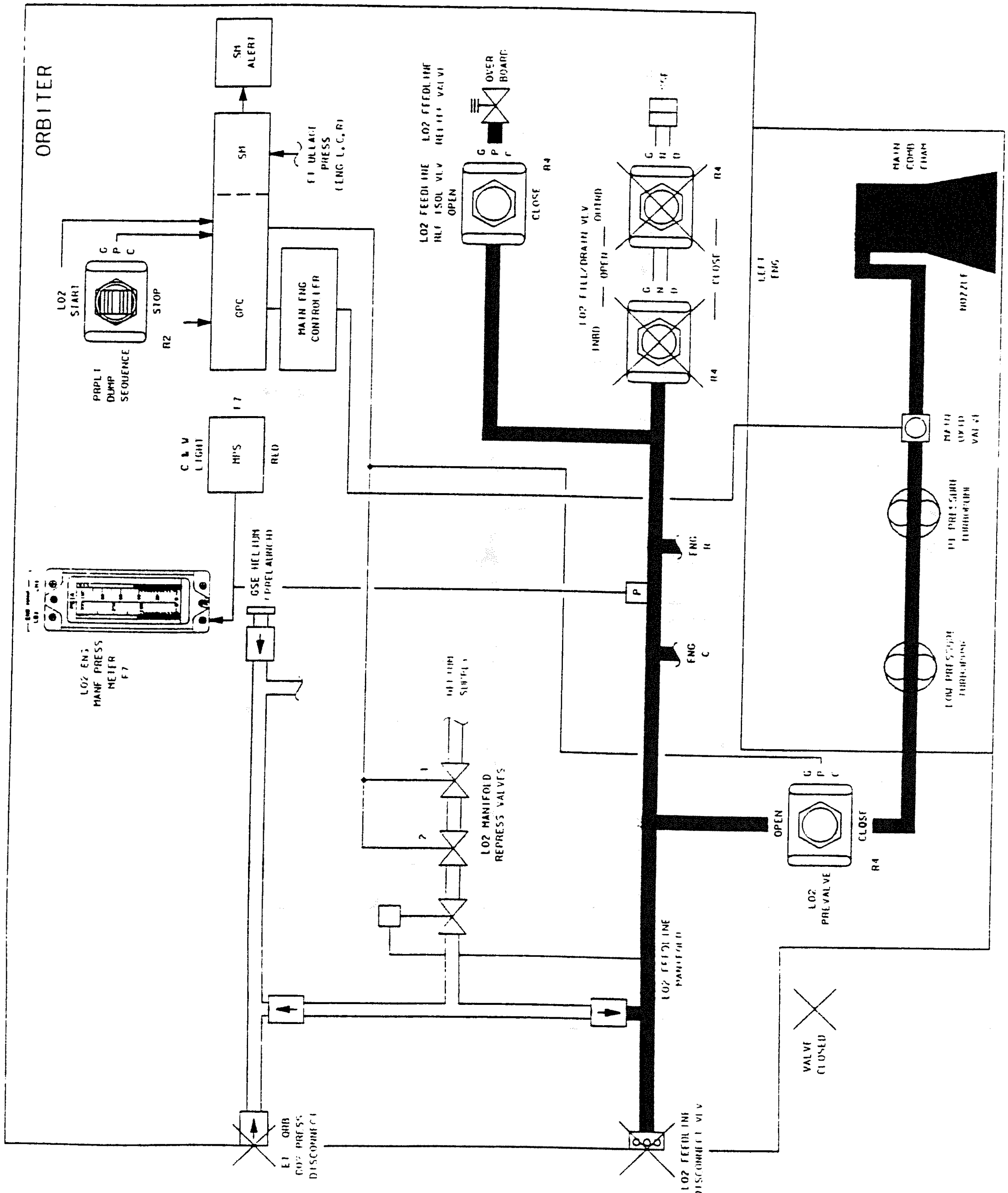


Figure 3.5-3.- MPS LO2 dump subsystem flow during dump.

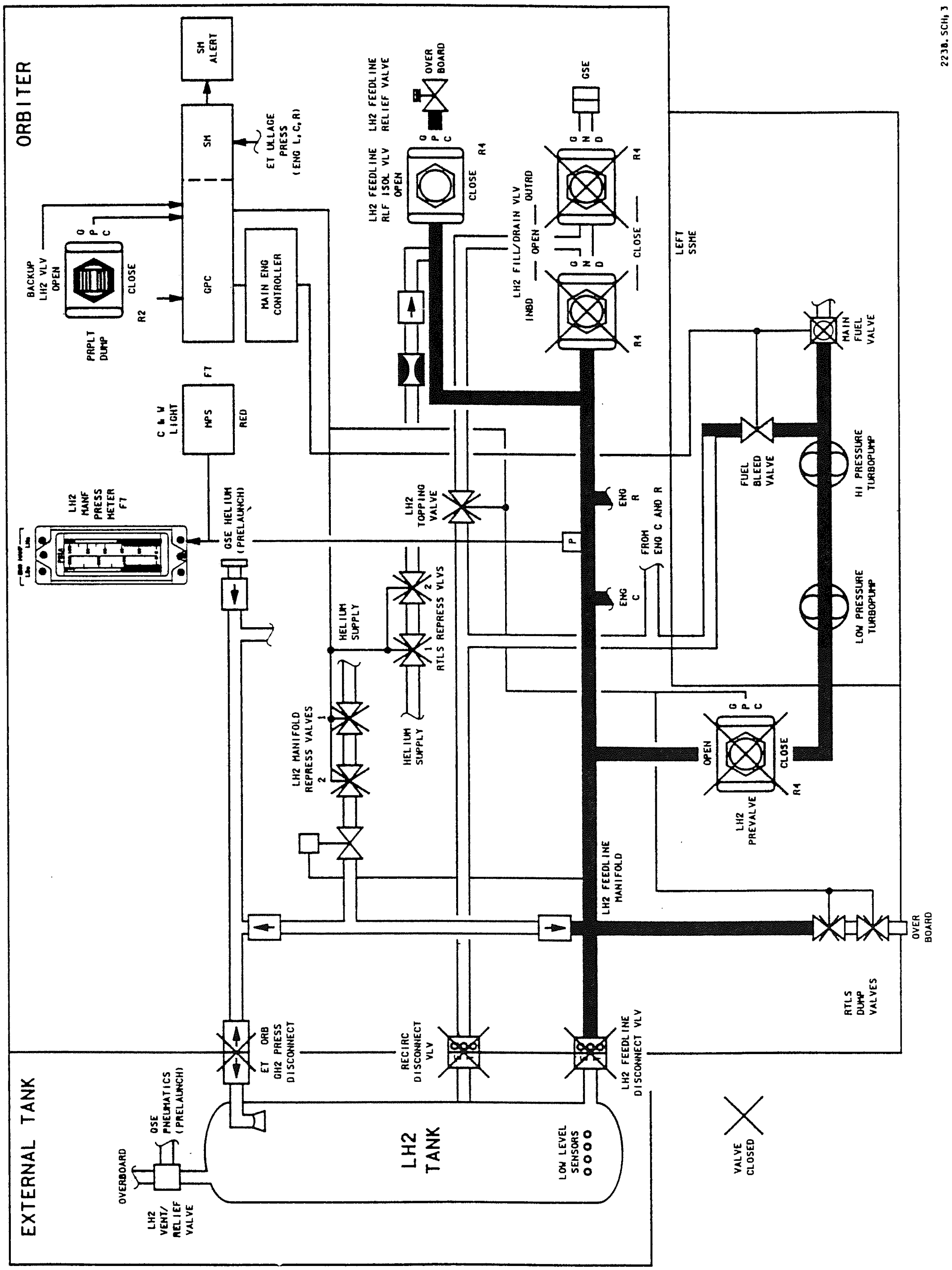


Figure 3.5-4.- MPS LH2 dump subsystem before dump.

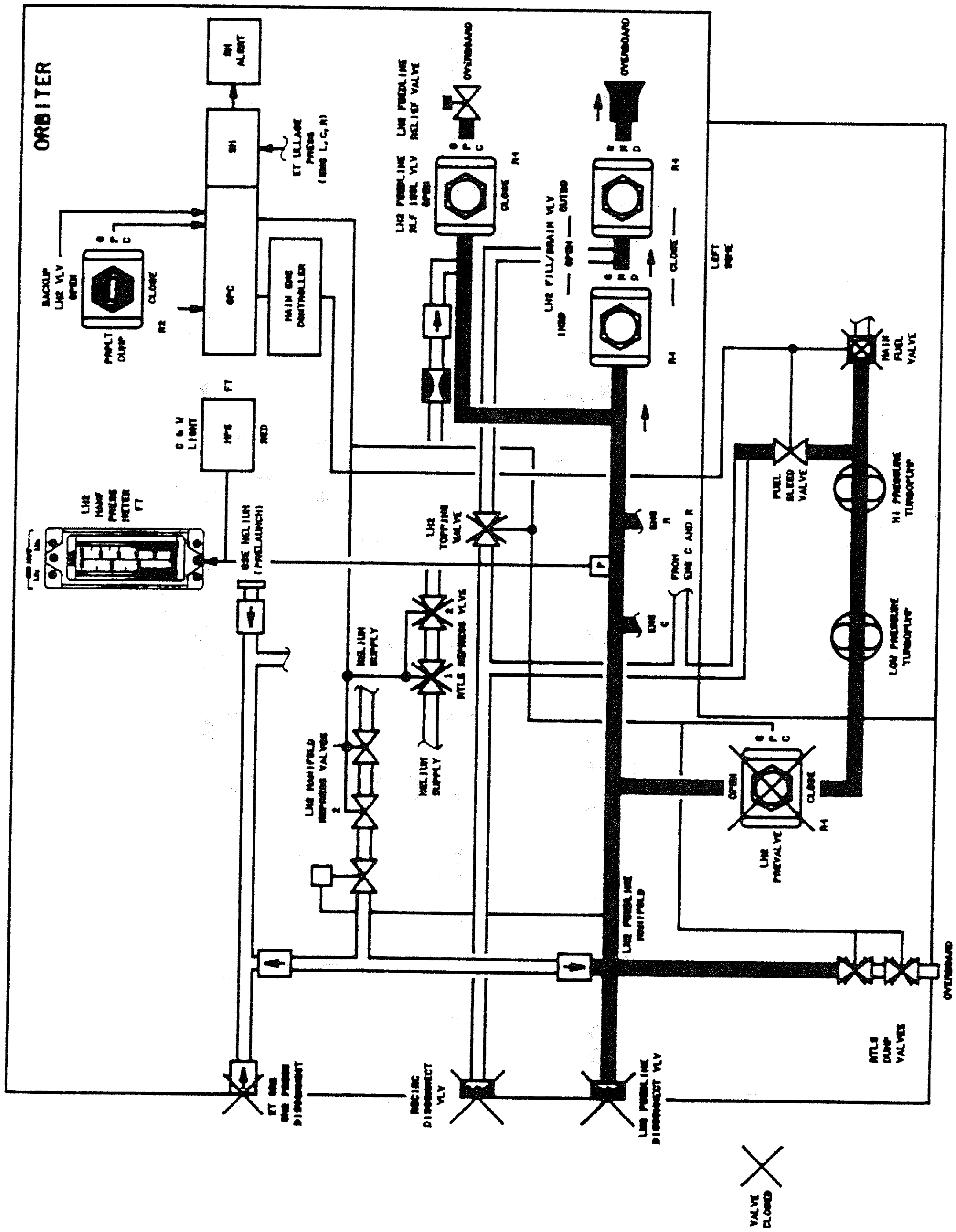


Figure 3.5-5.- MPS LH_2 dump subsystem flow from DUMP START to DUMP START +6 seconds.

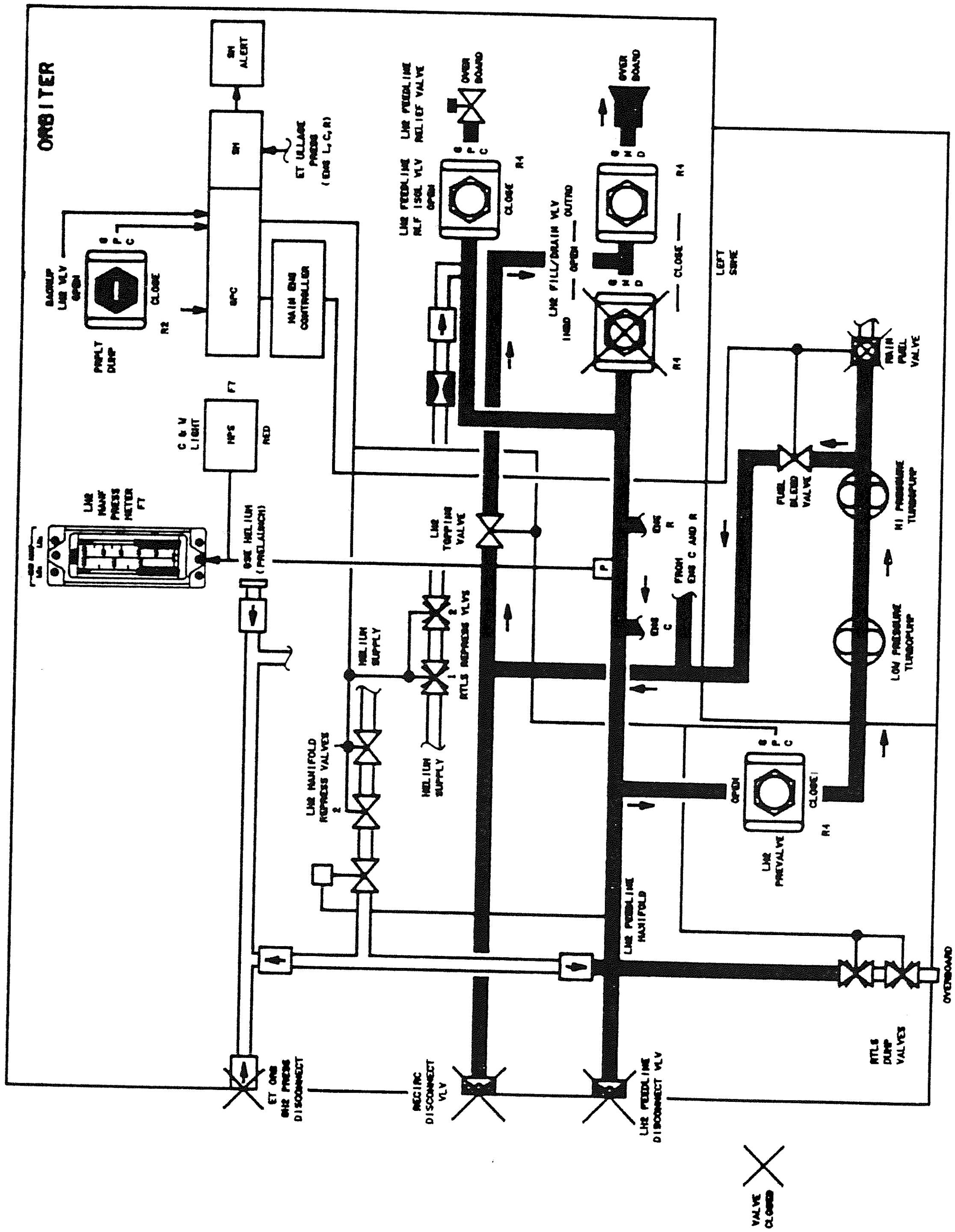


Figure 3.5-6.- MPS LH_2 dump subsystem flow after DUMP START +6 seconds.

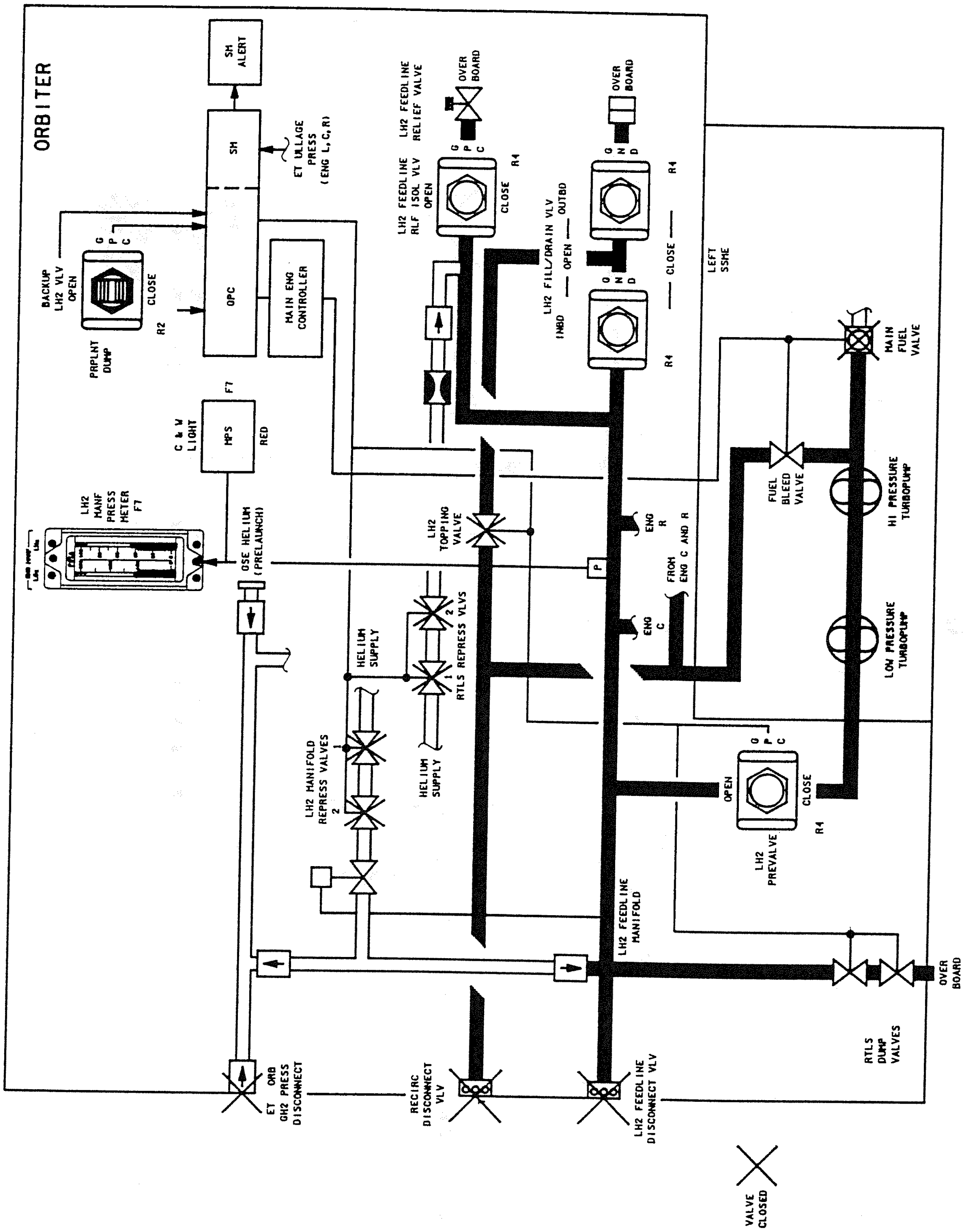


Figure 3.5-7.- MPS LH2 dump subsystem flow during dump - RTLS.

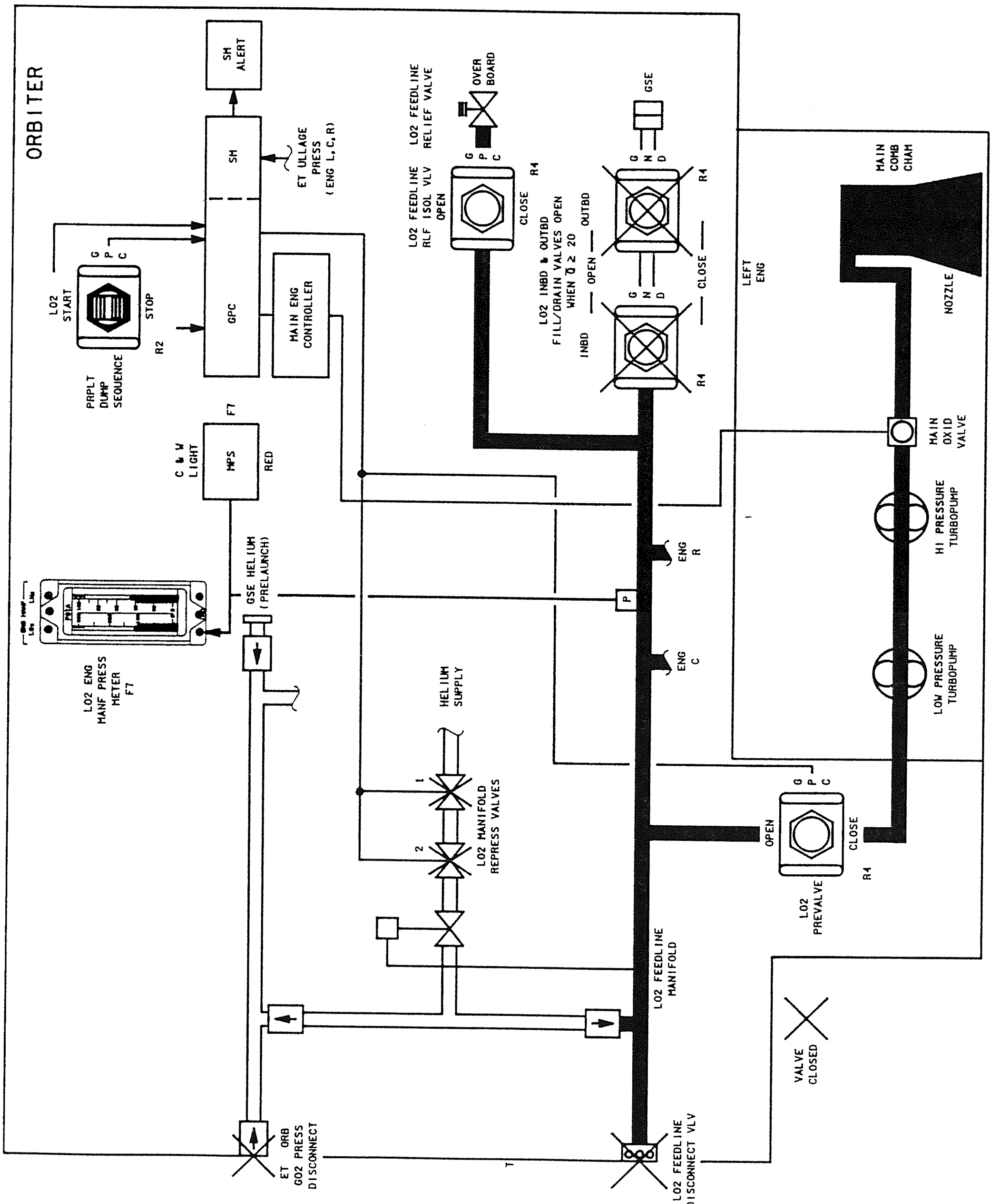
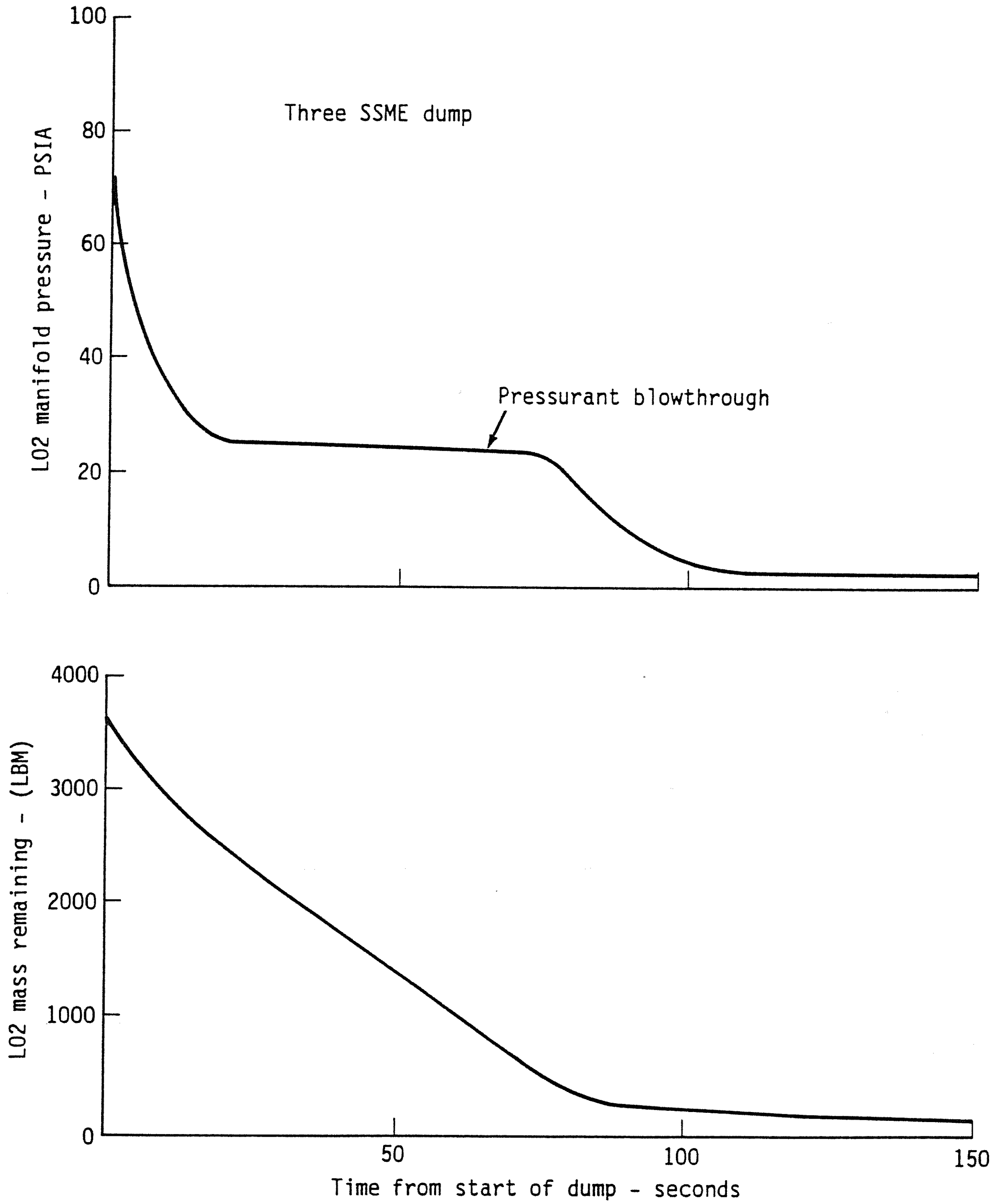


Figure 3.5-8.- MPS LO2 dump subsystem flow during dump - RTLS.



Note: 270 lb of LO_2 leaked through HPOTP seal in 120 seconds to start of dump (normal insertion).

Figure 3.5-9.- MPS LO_2 feed system pressure and mass during MM 104 dump (normal, PTM, ATO, AOA mission).

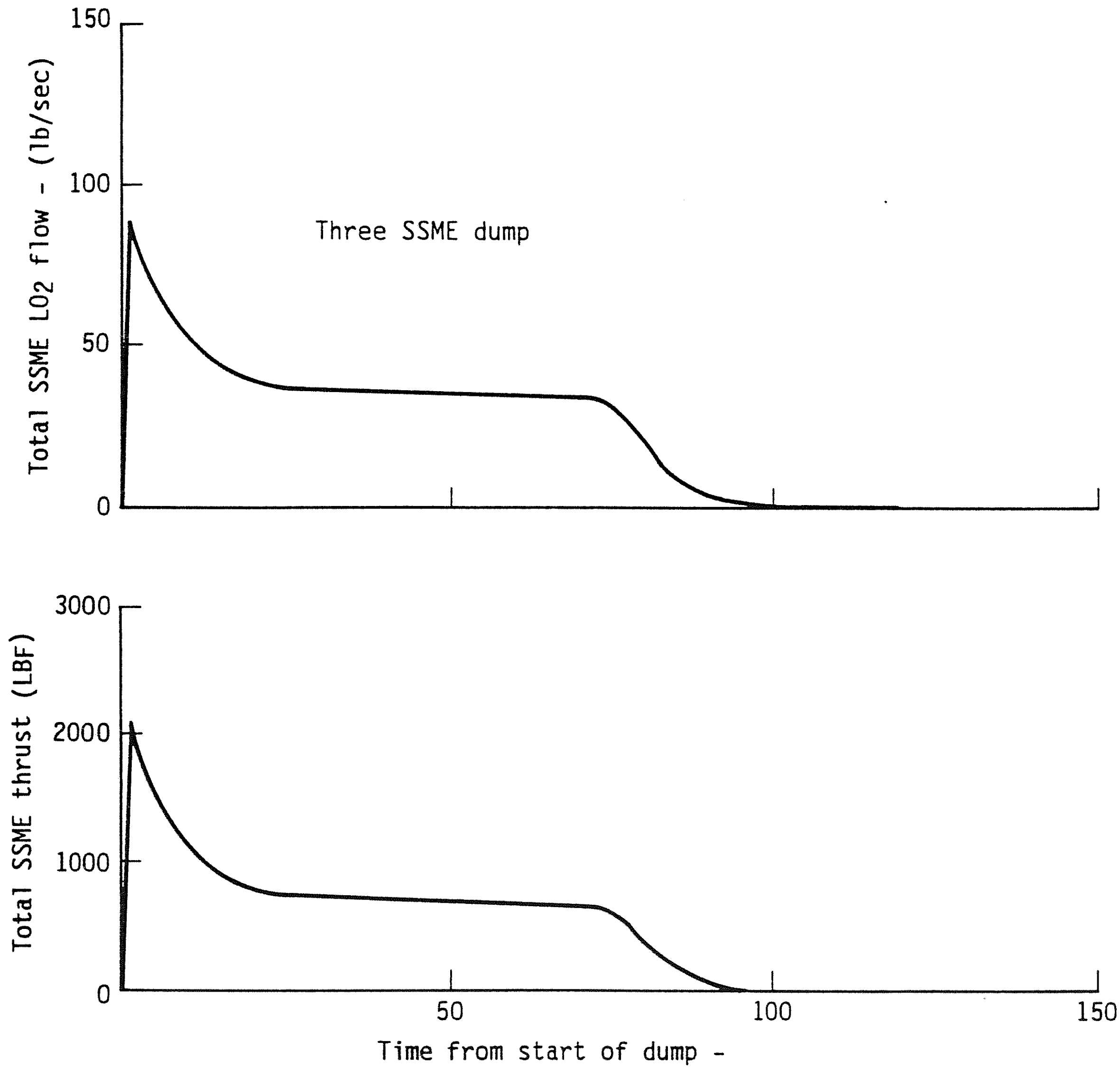
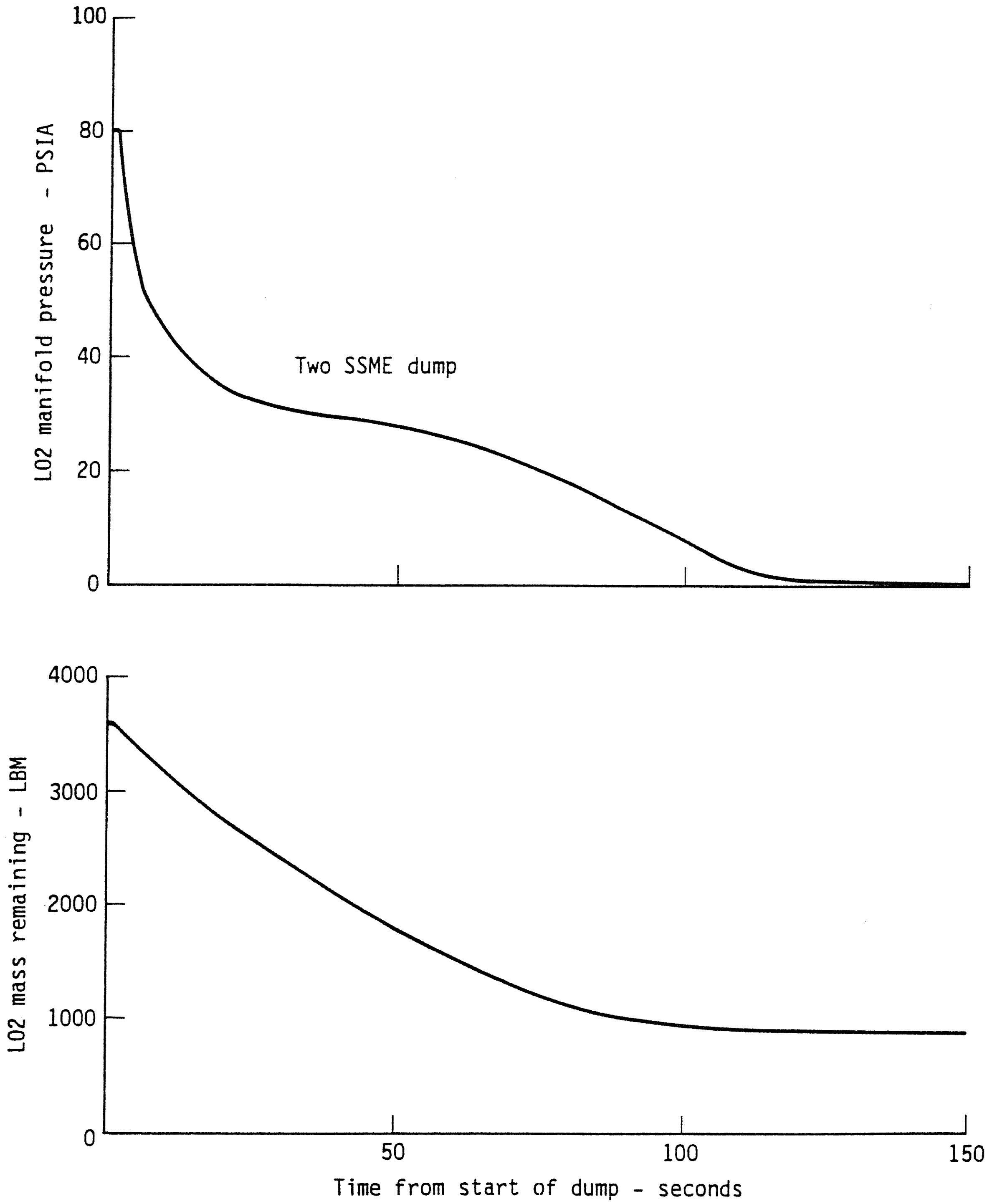


Figure 3.5-10.- MPS LO2 flowrate and thrust during MM 104 dump (normal, PTM, ATO, AOA mission).



Note: 270 lb of LO2 leaked through HPOTP seal in 120 seconds from MECO to start of dump (normal insertion).

Figure 3.5-11.- MPS LO2 feed system pressure and mass during MM 104 dump (normal, PTM, ATO, AOA mission).

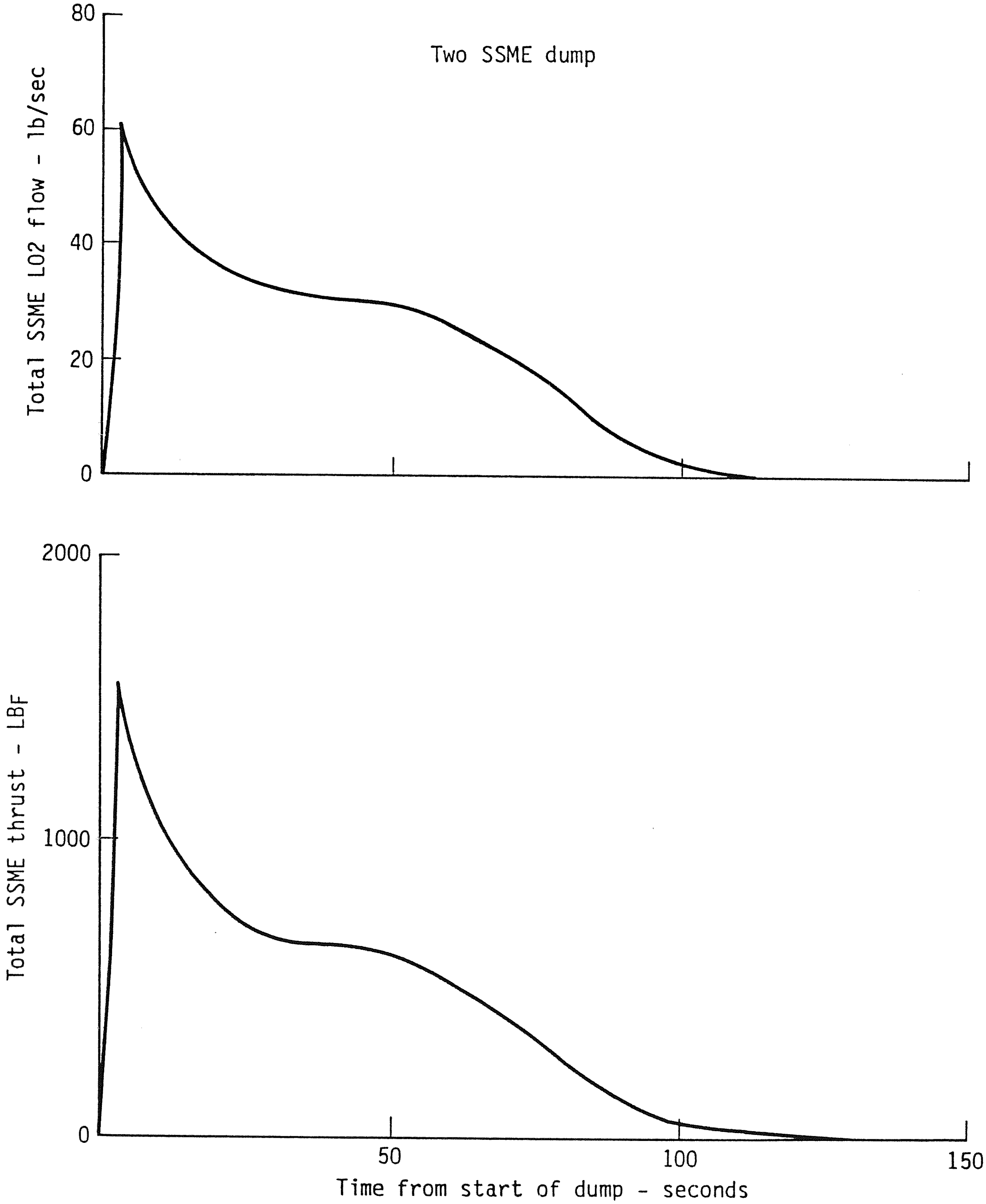


Figure 3.5-12.- MPS LO2 flowrate and thrust during MM 104 dump (normal, PTM, ATO, AOA mission).

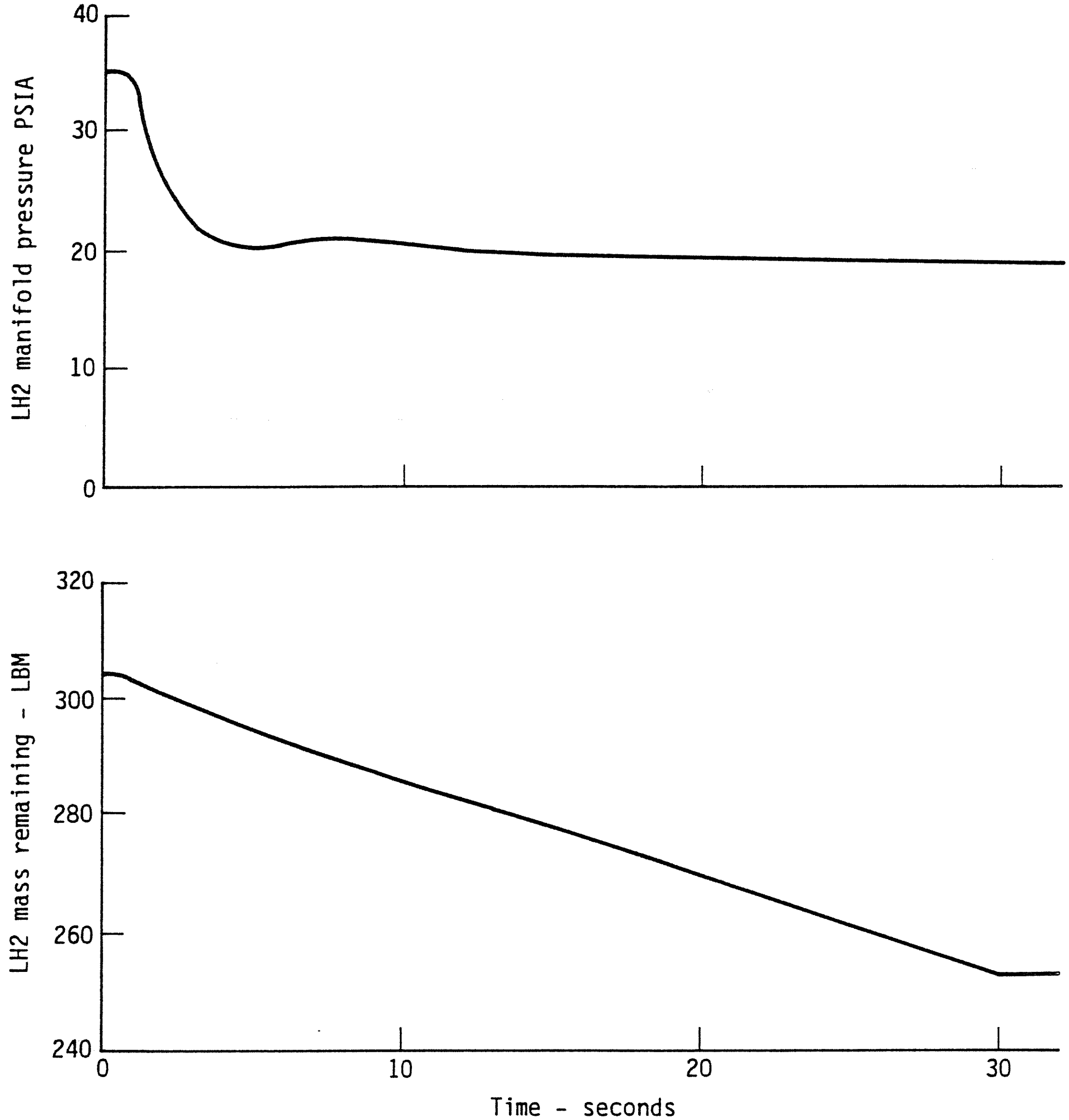


Figure 3.5-13.- MPS LH2 feed system pressure and mass during 30-second feedline venting (MM104).

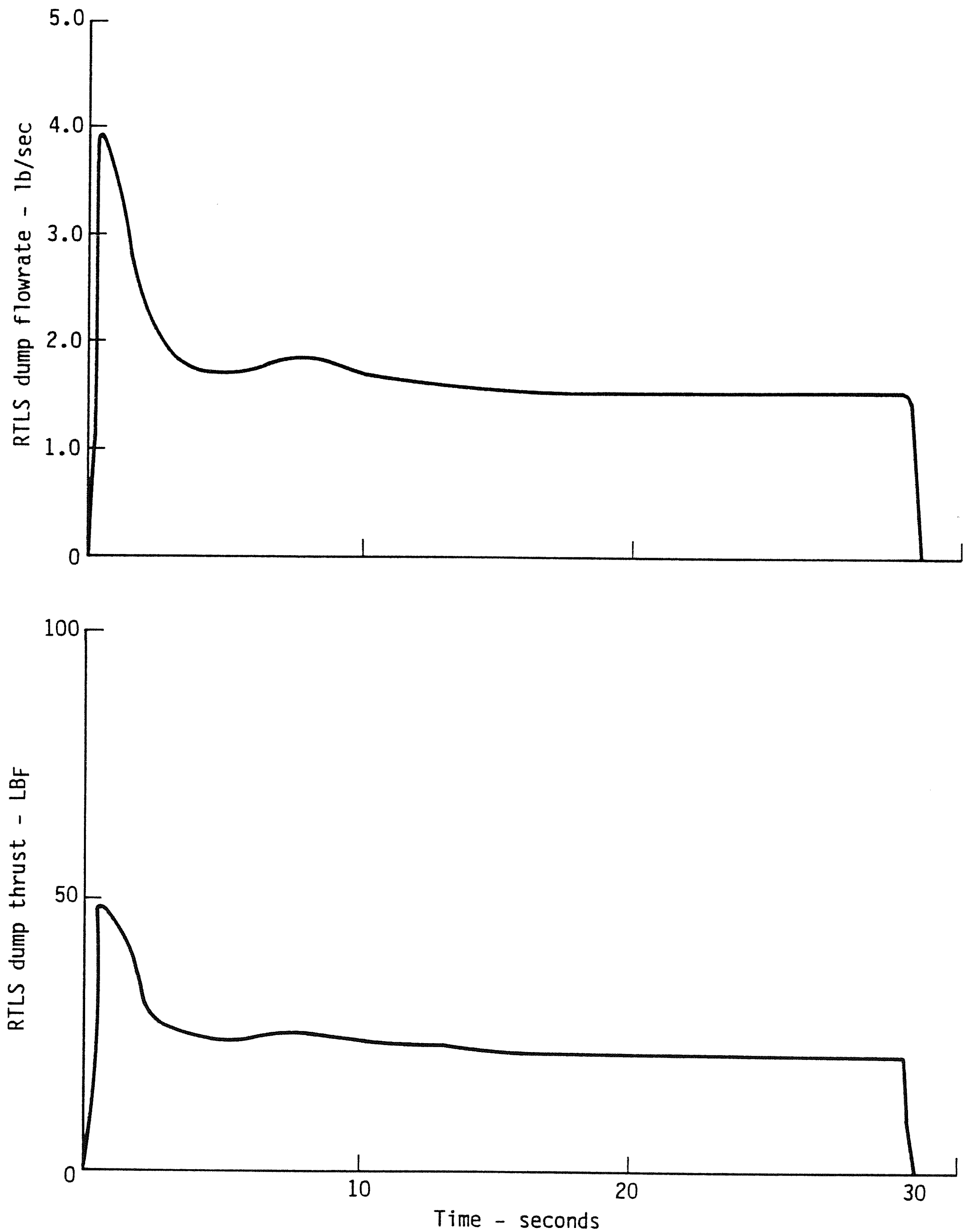


Figure 3.5-14.- RTLS line flowrate and thrust during 20-second feedline venting (MM 104).

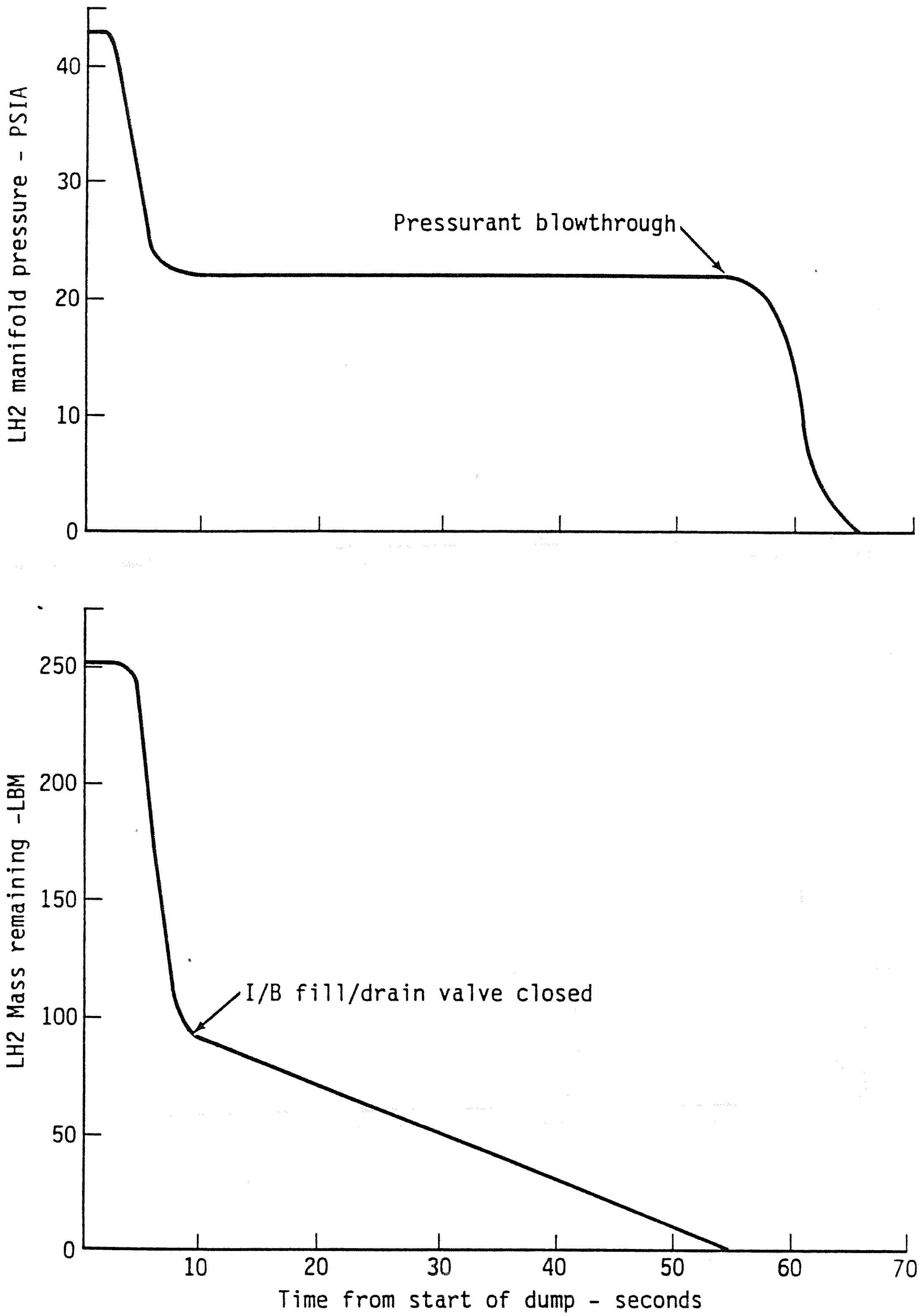


Figure 3.5-15.- MPS LH2 feed system pressure and mass during MM 1204 dump (normal, PTM, ATO, AOA mission).

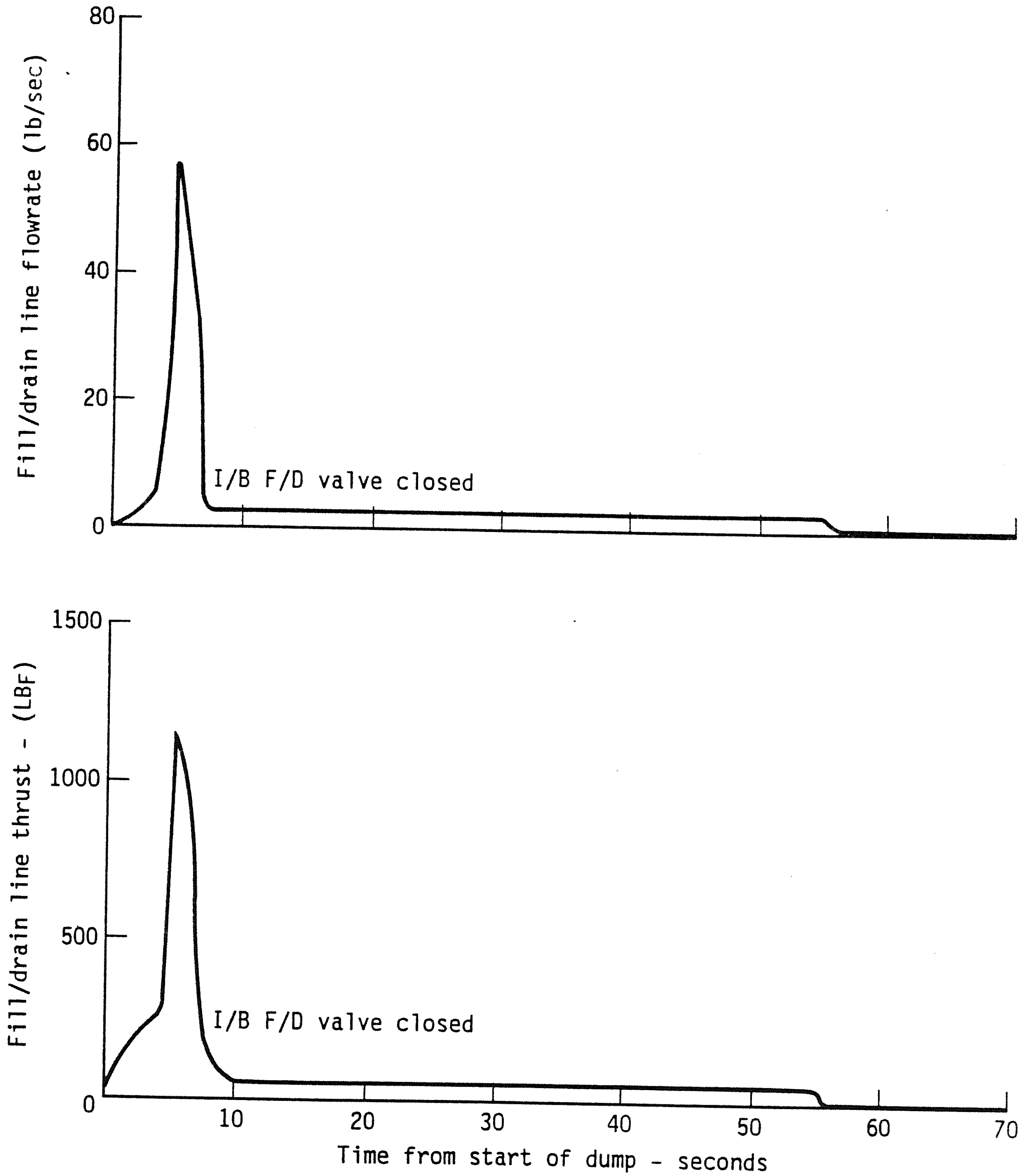


Figure 3.5-16.- LH_2 fill/drain flowrate and thrust during MM 104 dump (normal, PTM, ATO, AOA mission).

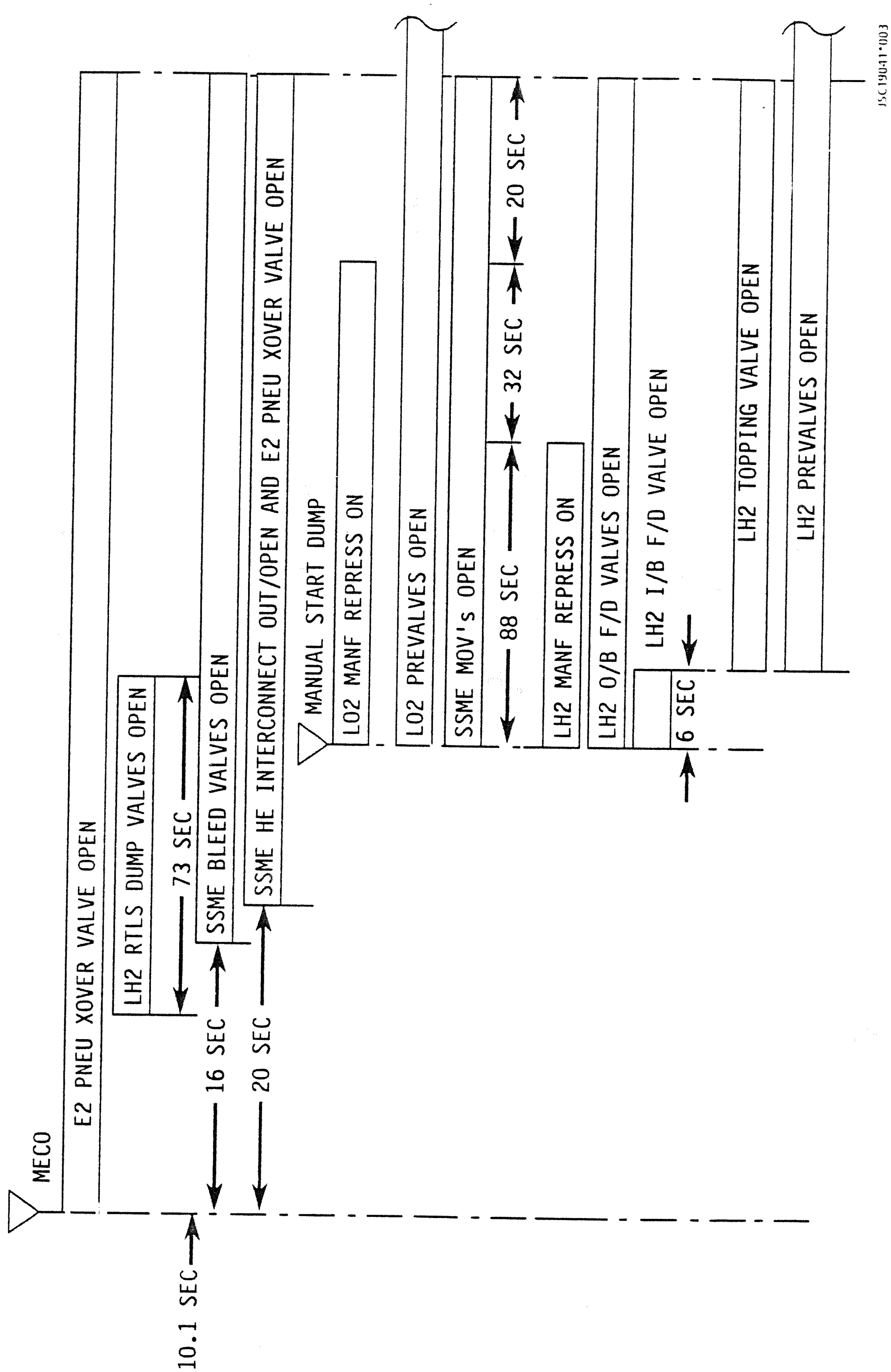


Figure 3.5-17.- Manual MPS dump sequence.

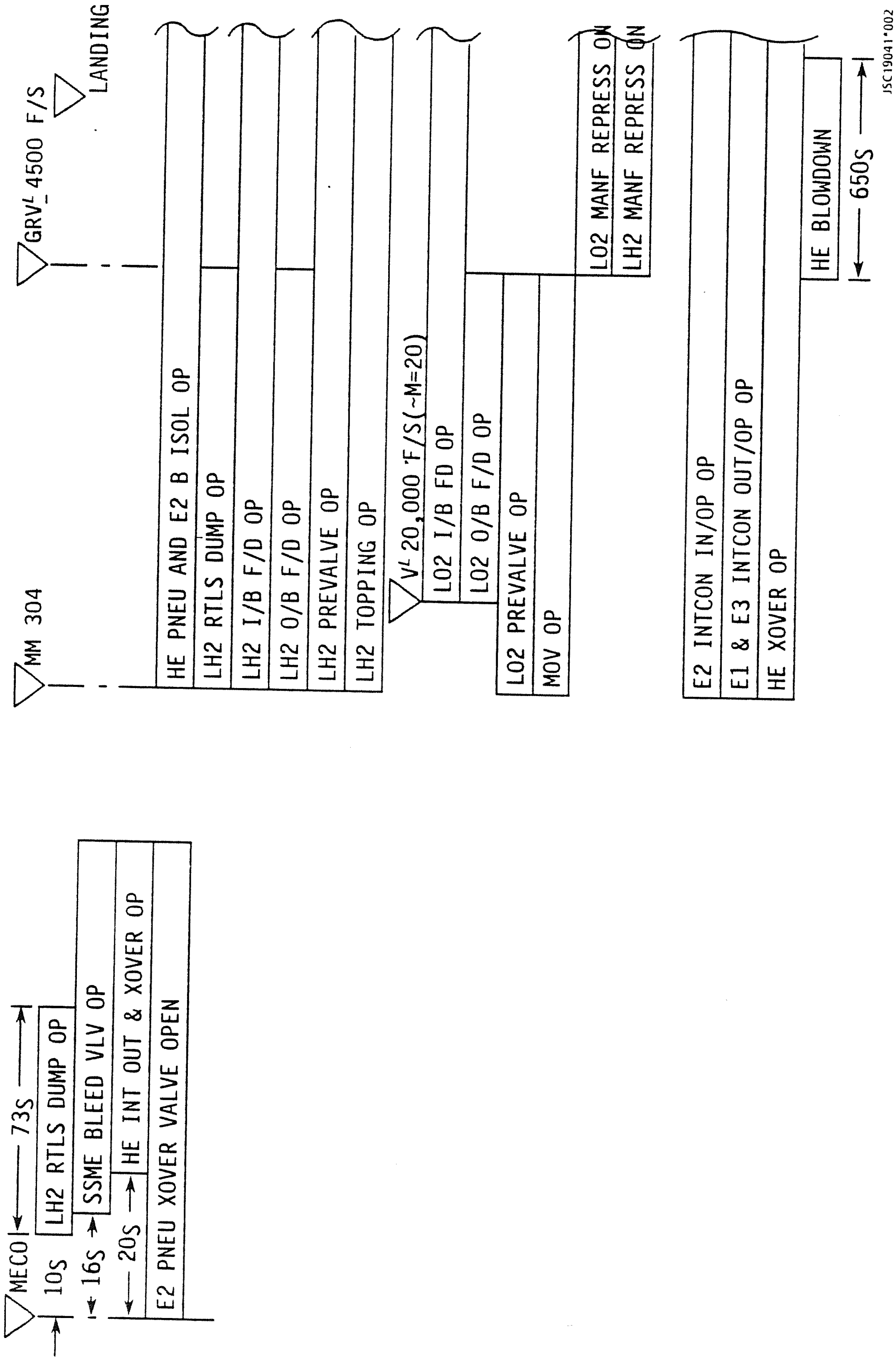


Figure 3.5-18.- TAL MPS dump sequence.

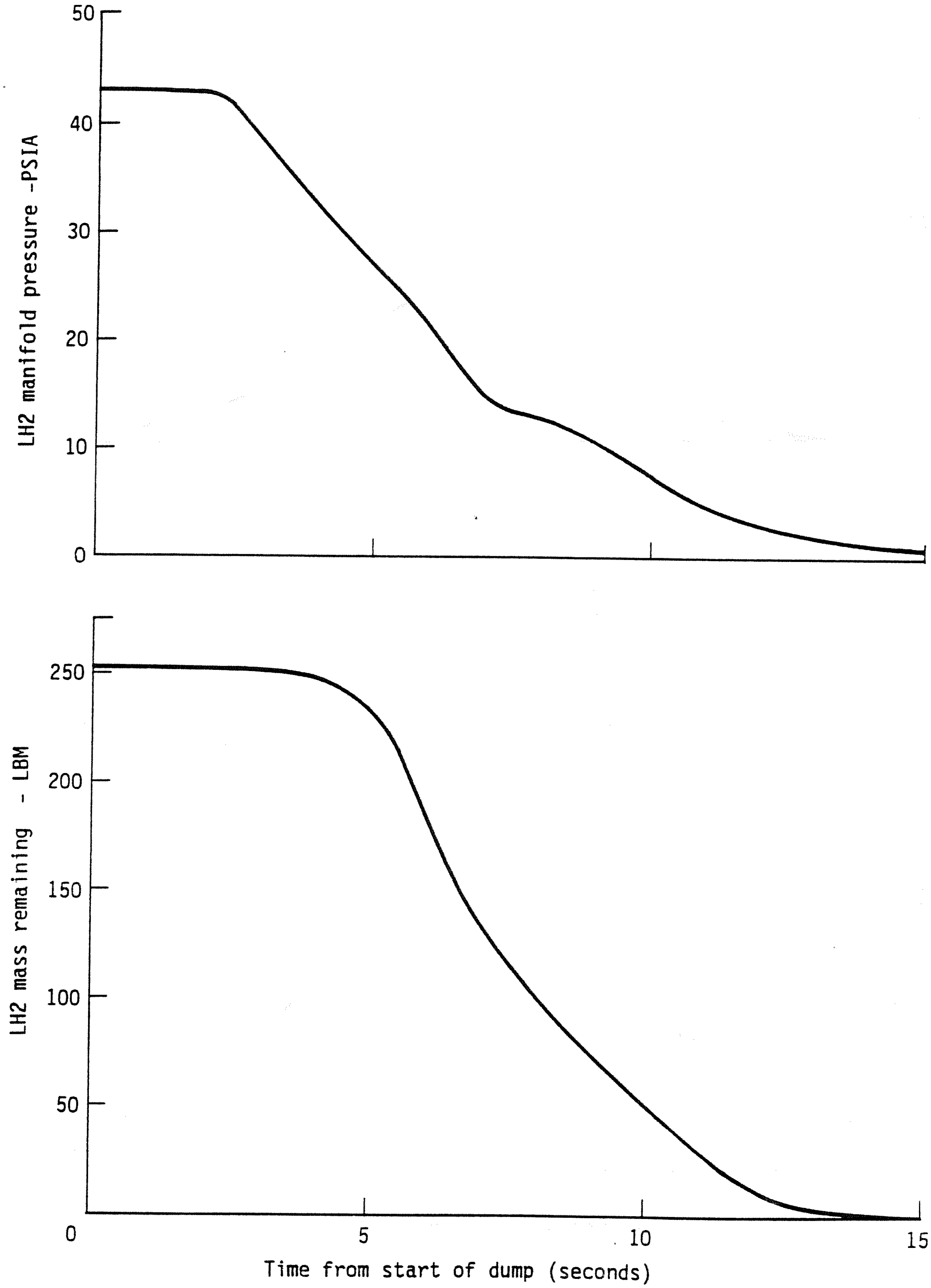


Figure 3.5-19.- MPS LH_2 feed system pressure and mass during TAL dump (MM 304) or RTLS dump (MM 602).

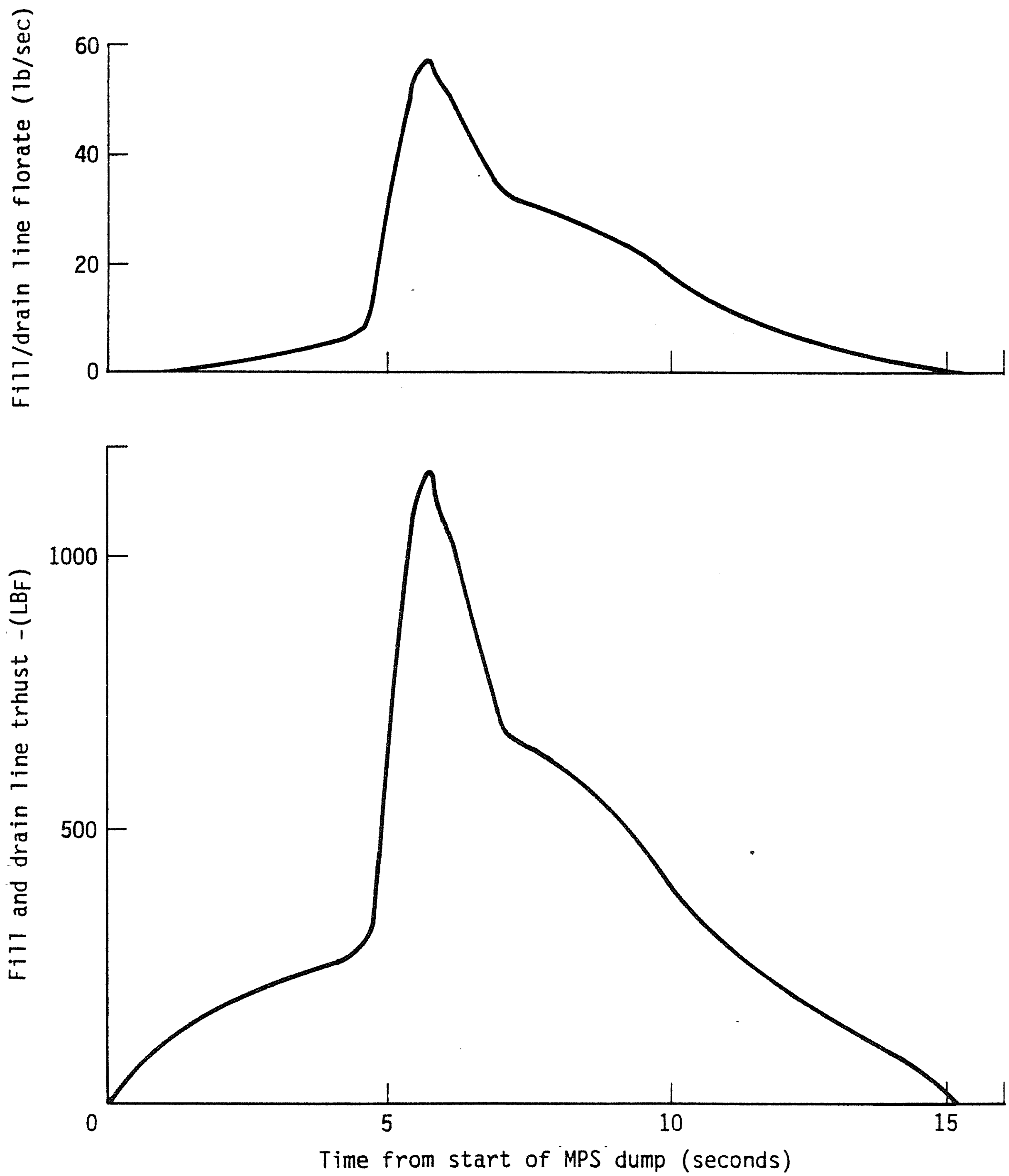


Figure 3.5-20.- LH_2 fill/drain flowrate and thrust during TAL dump (MM 304) or RTLS dump (MM 602).

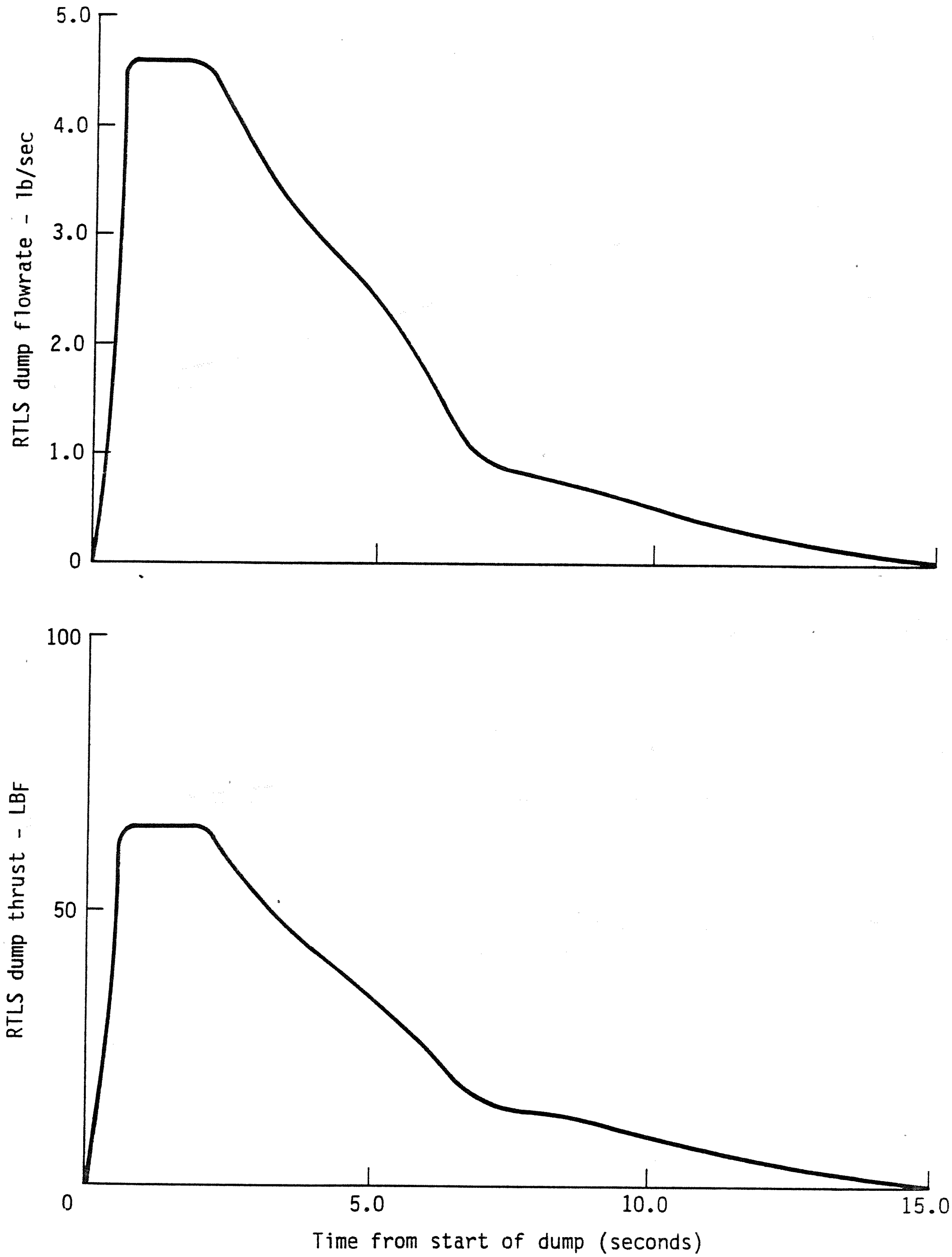


Figure 3.5-21.- LH_2 RTLS line flowrate and thrust during TAL dump (MM 304) or RTLS dump (MM 602).

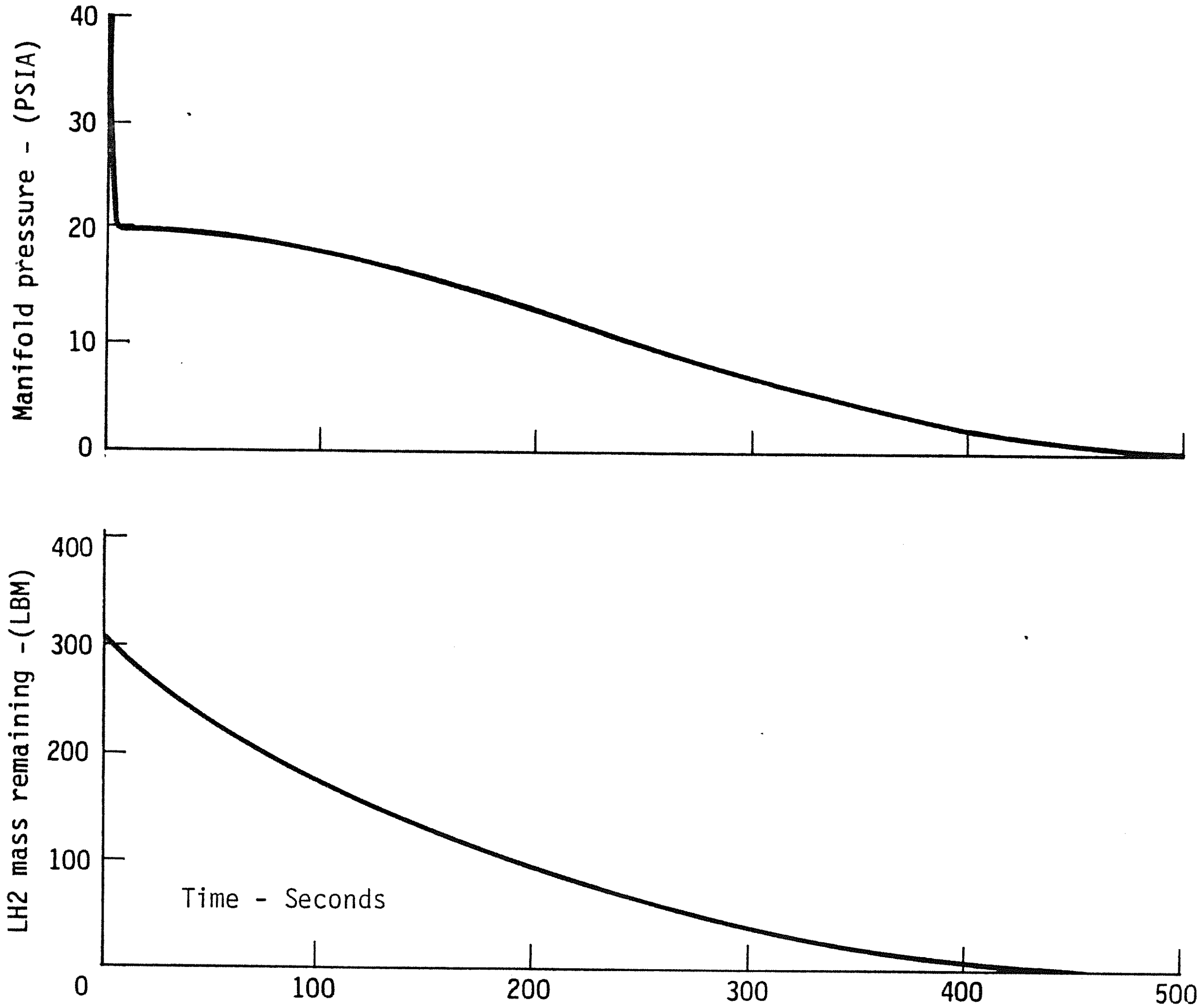


Figure 3.5-22- MPS LH2 feed system pressure and mass during TAL dump (MM 304) with F/D valve failed closed.

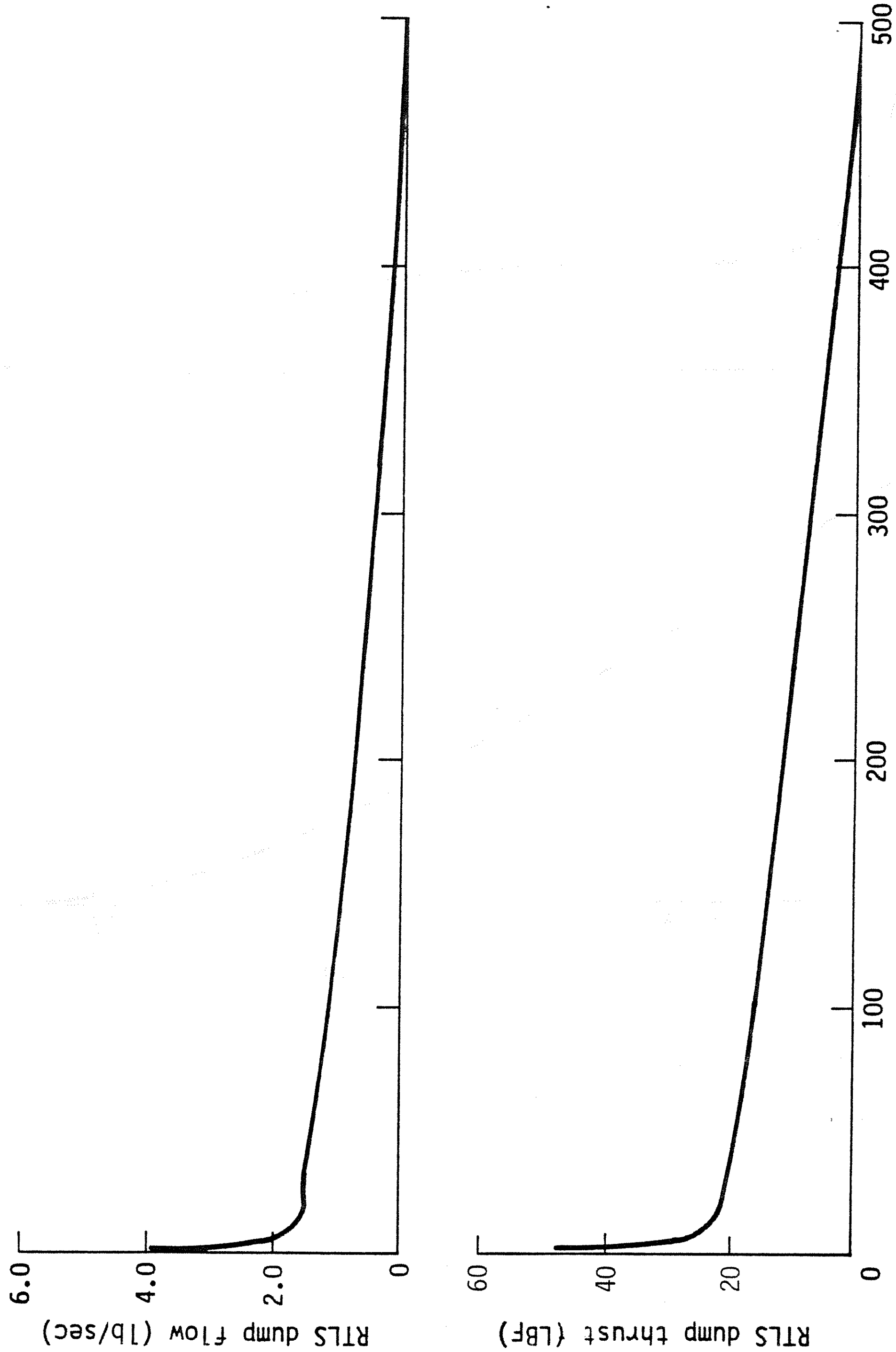


Figure 3.5-23- RTLS line flowrate and thrust during TAL dump (MM 304) with F/D valve failed closed.

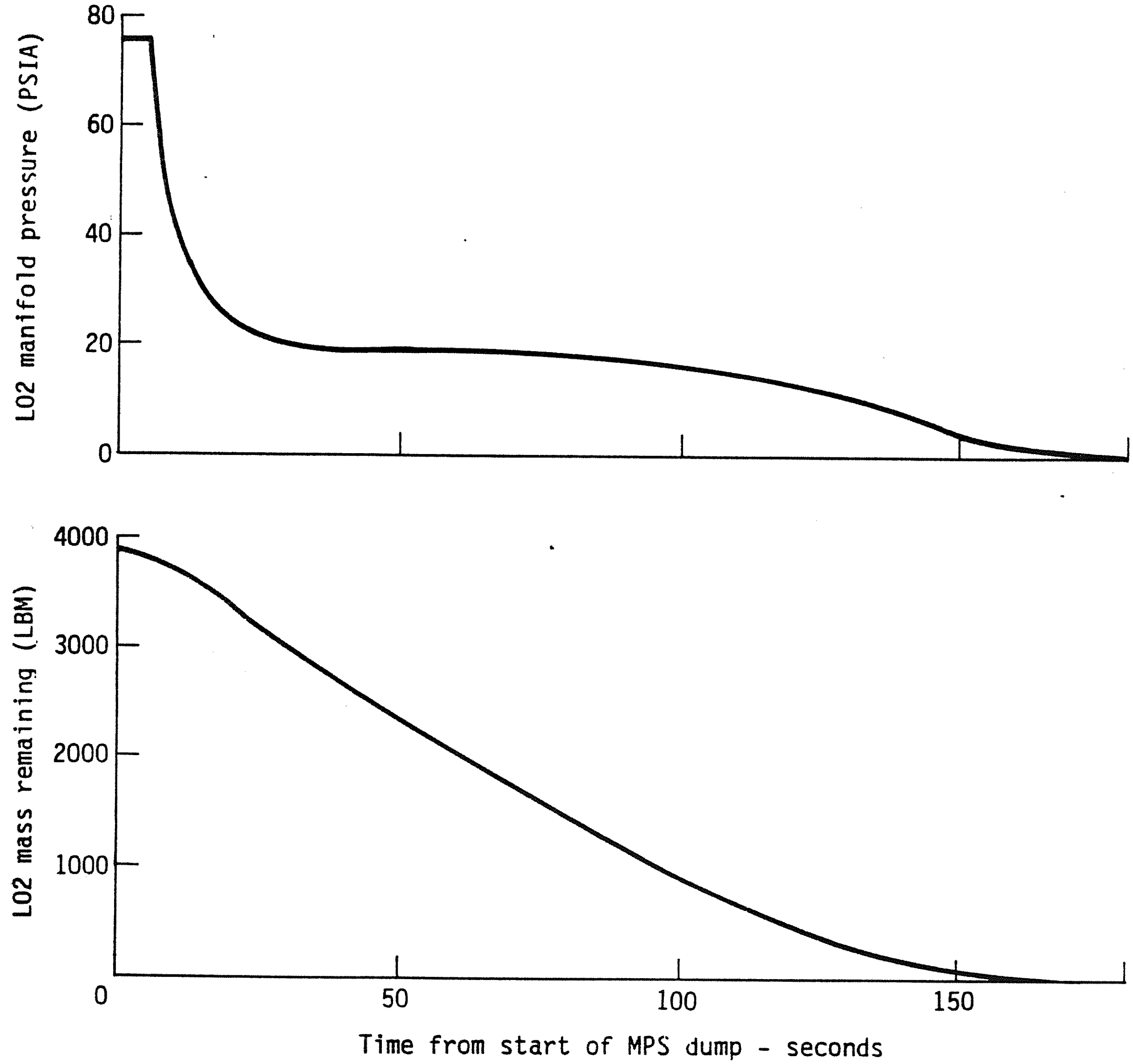


Figure 3.5-24.- MPS LO2 feed system pressure and mass during TAL dump (MM 304) (three SSME dump).

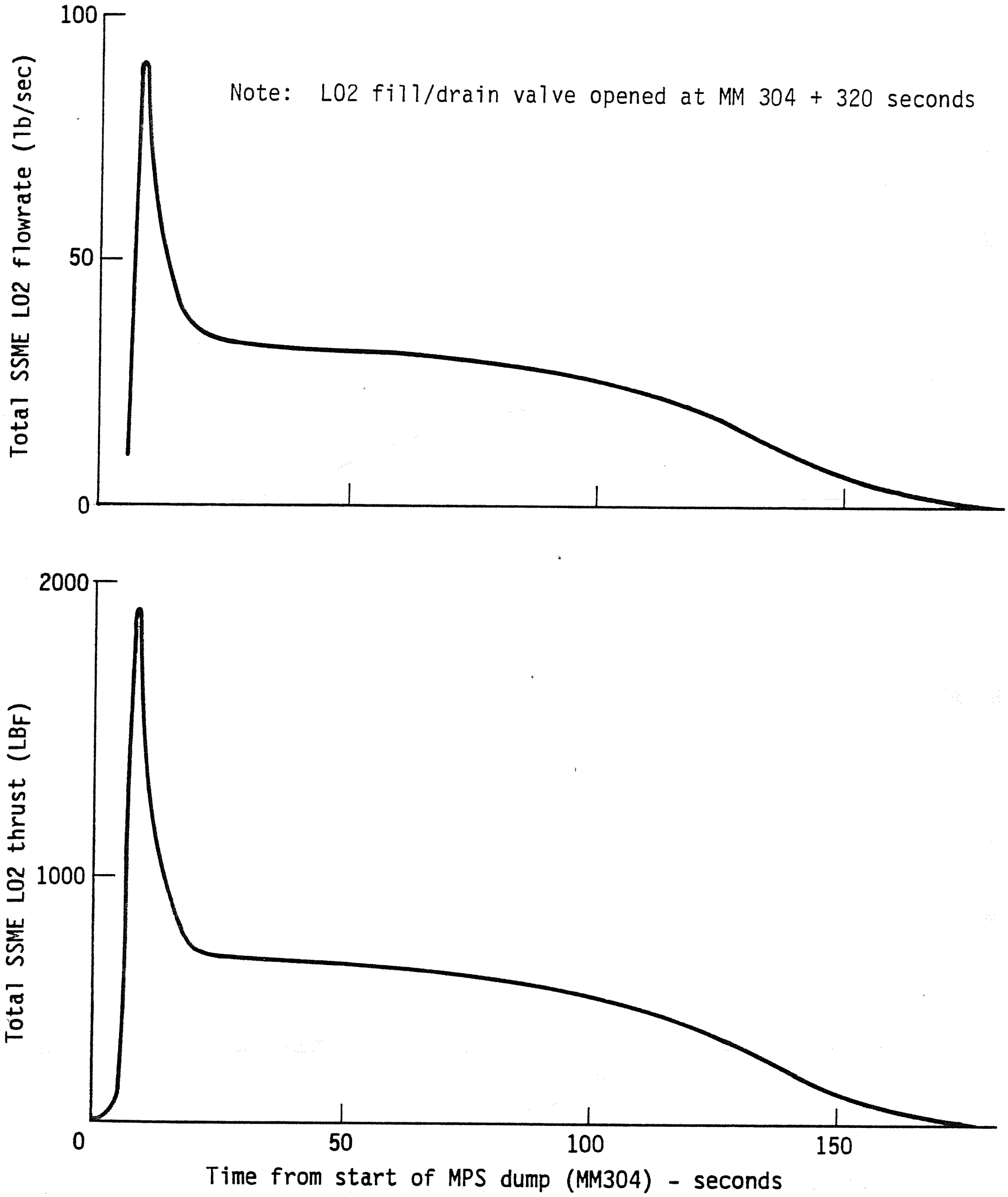


Figure 3.5-25.- SSME LO2 flowrate and thrust during TAL dump (MM 304) (three SSME dump).

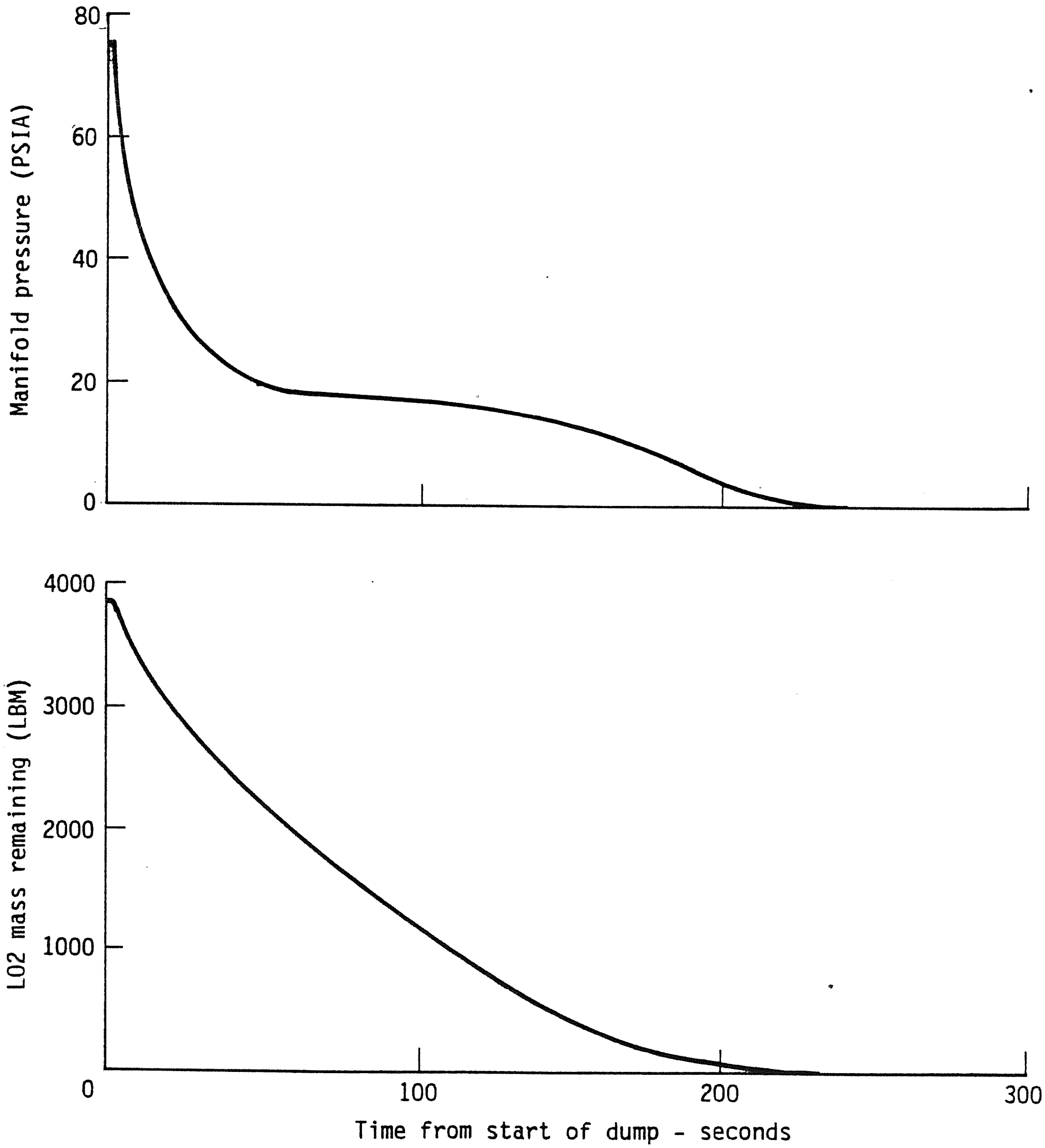


Figure 3.5-26.- MPS LO2 feed system pressure and mass during TAL dump (MM 304) (two SSME dump).

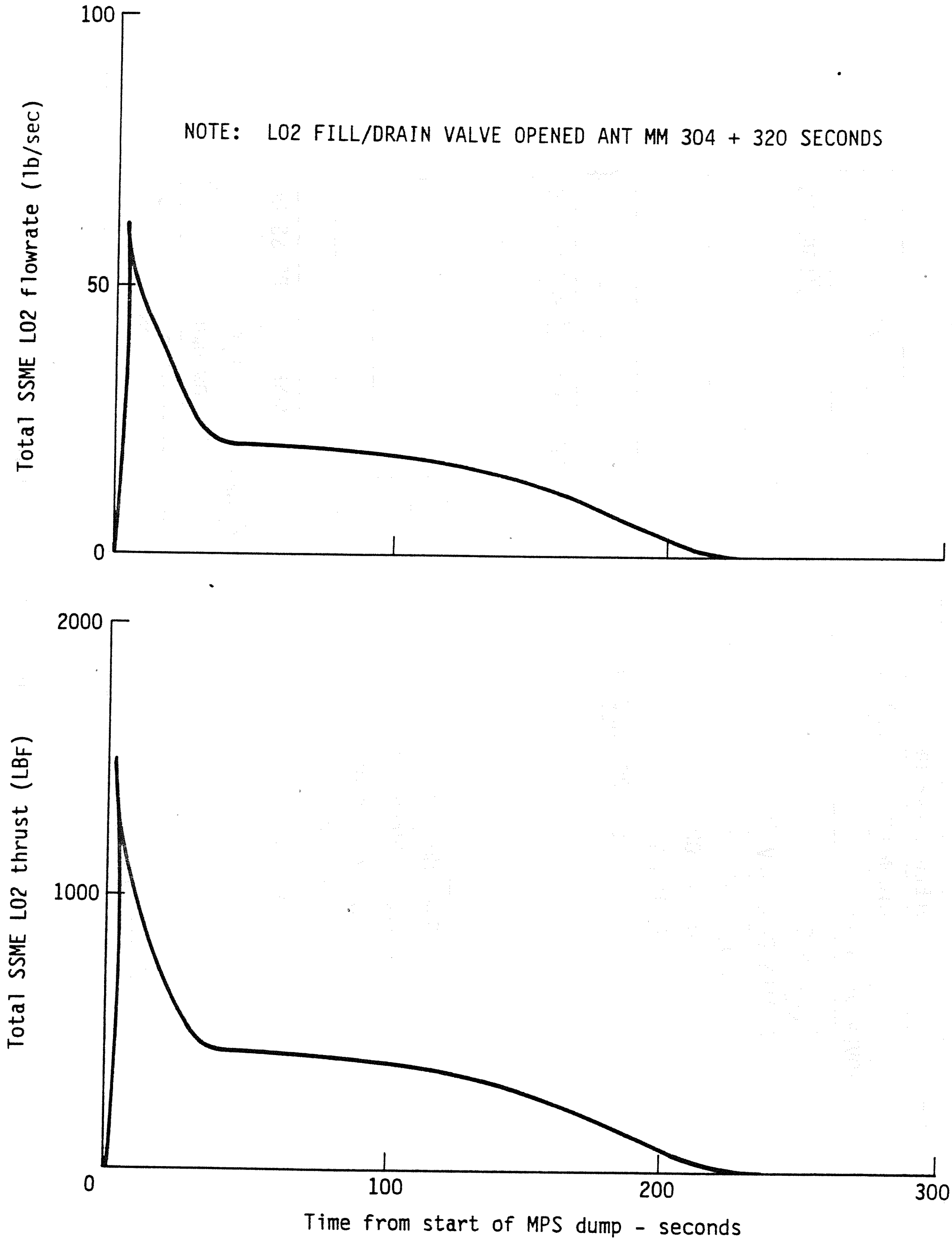


Figure 3.5-27.- SSME LO2 flowrate and thrust during TAL dump (MM 304)
(two SSME dump).

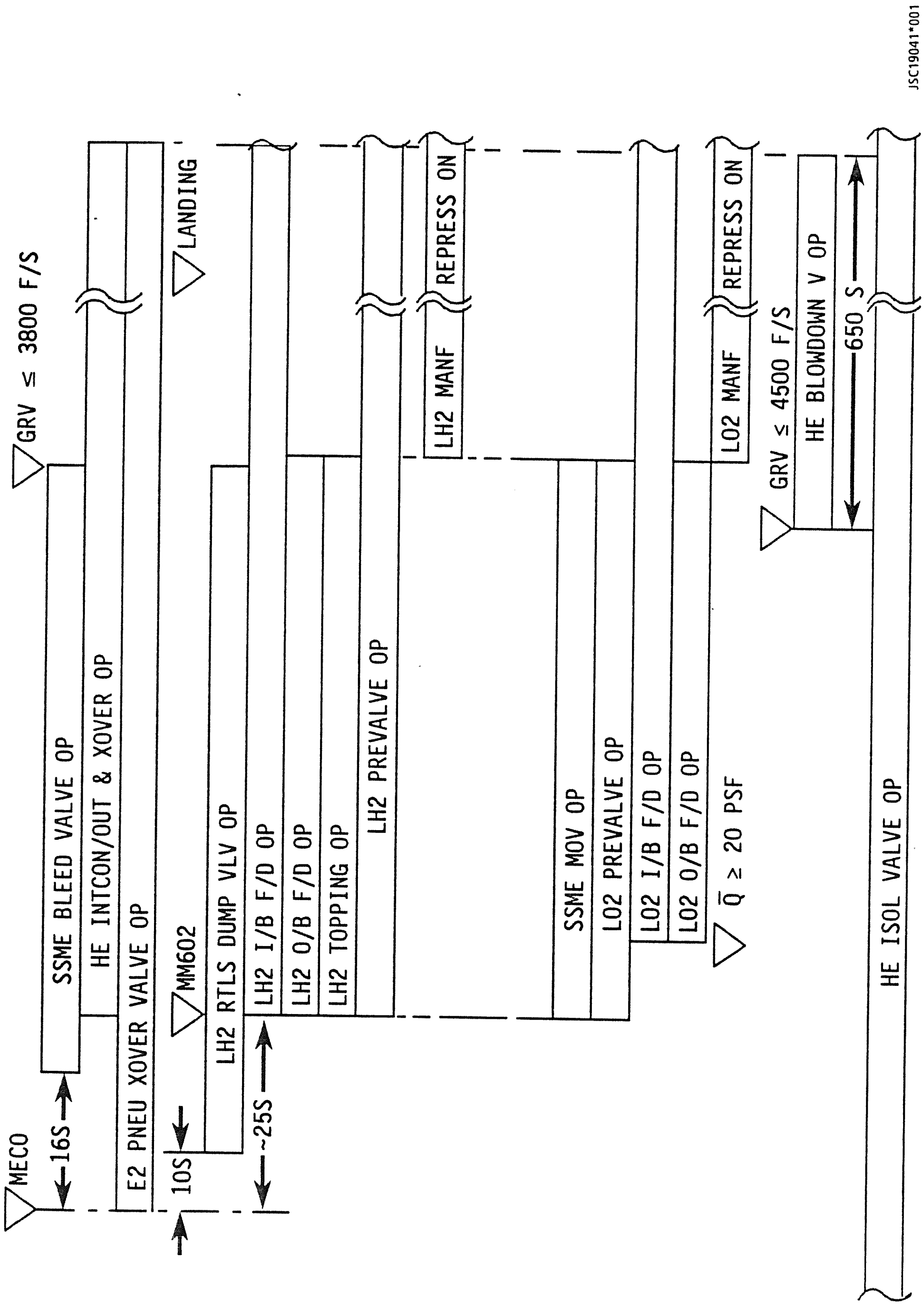


Figure 3.5-28.- RTLS and contingency dump.

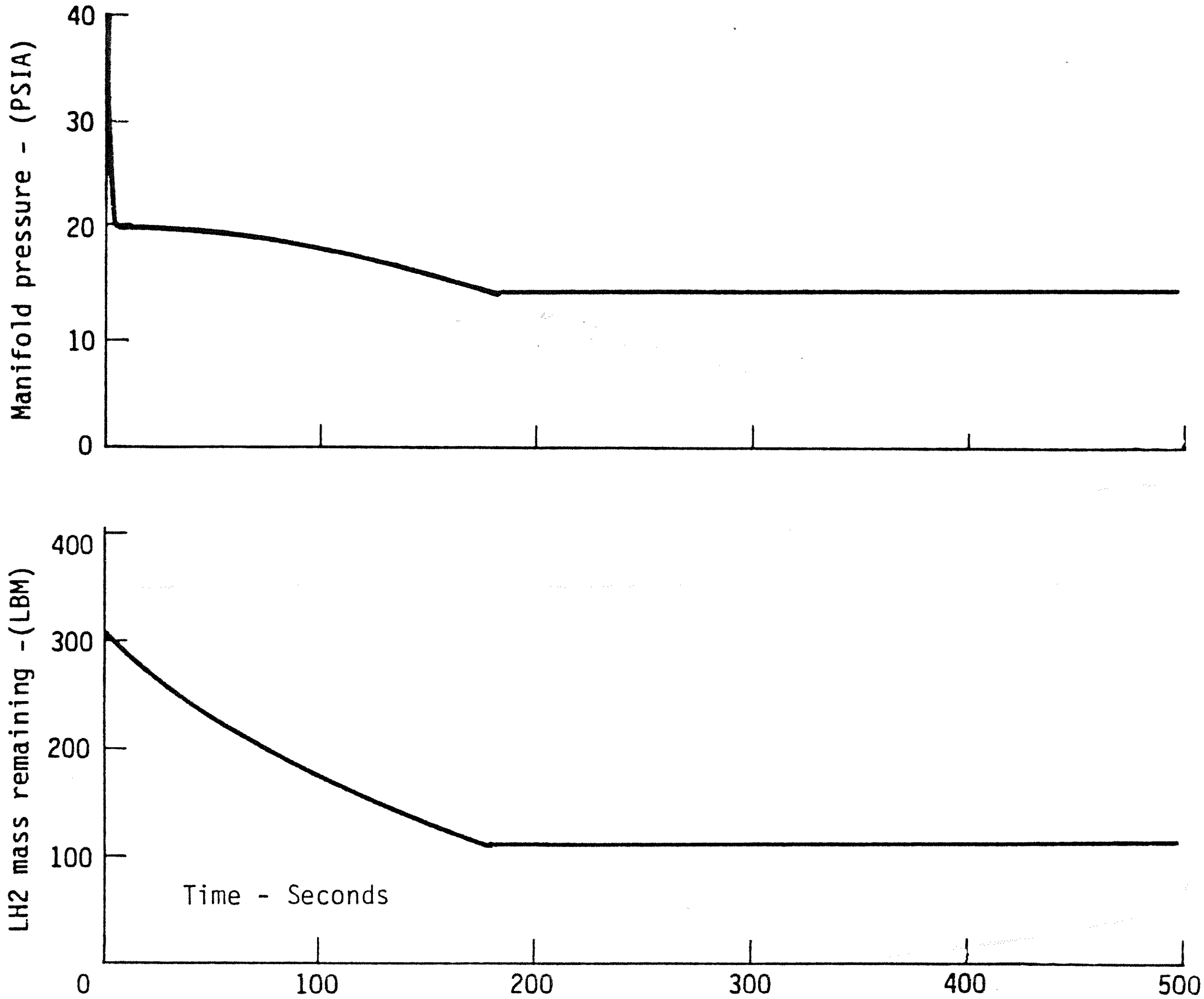


Figure 3.5-29.- MPS LH_2 feed system pressure and mass during RTLS dump (MM 602) with F/D valve failed closed.

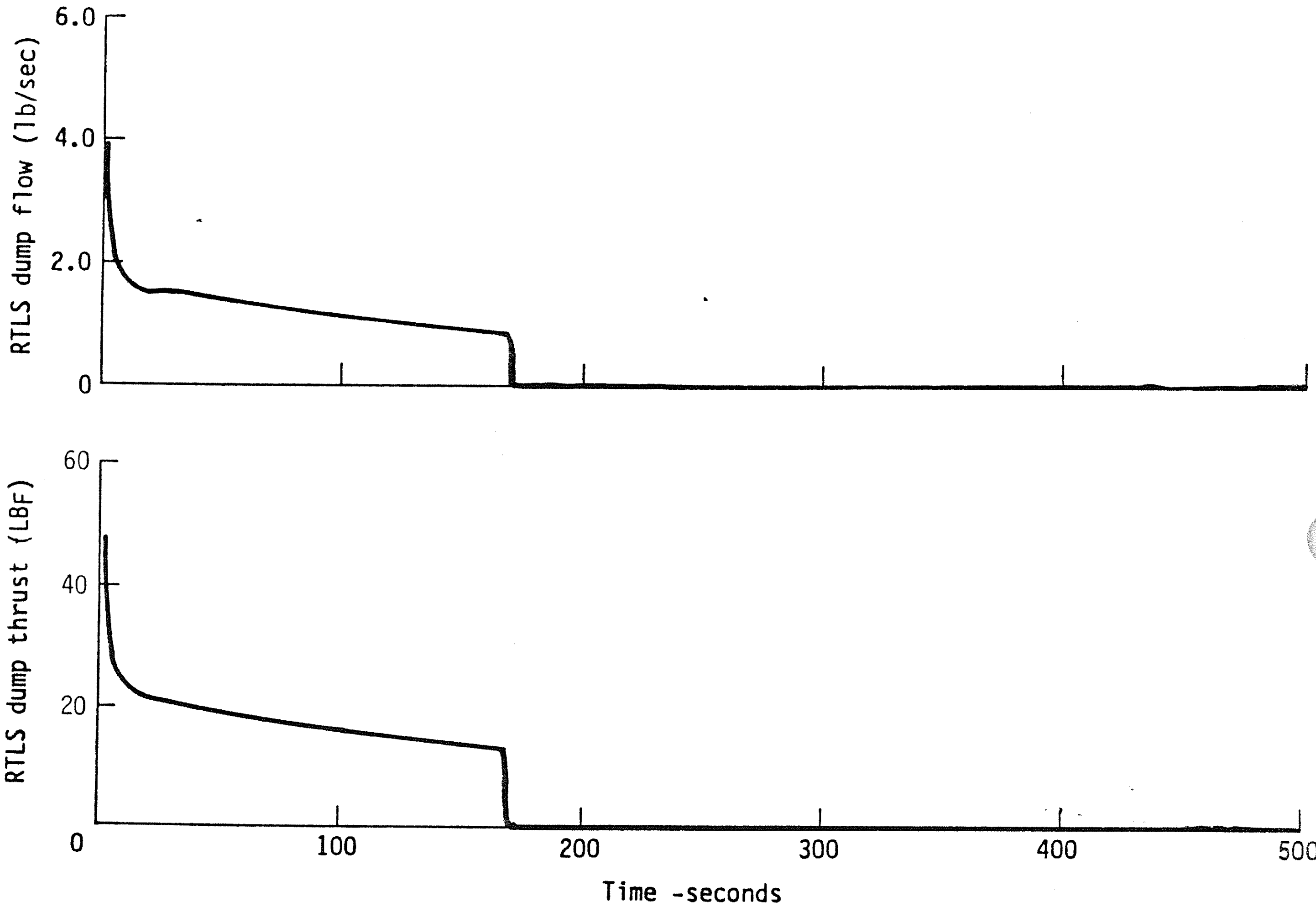


Figure 3.5-30.- RTLS line flowrate and thrust during RTLS dump (MM 602) with F/D valve failed closed.

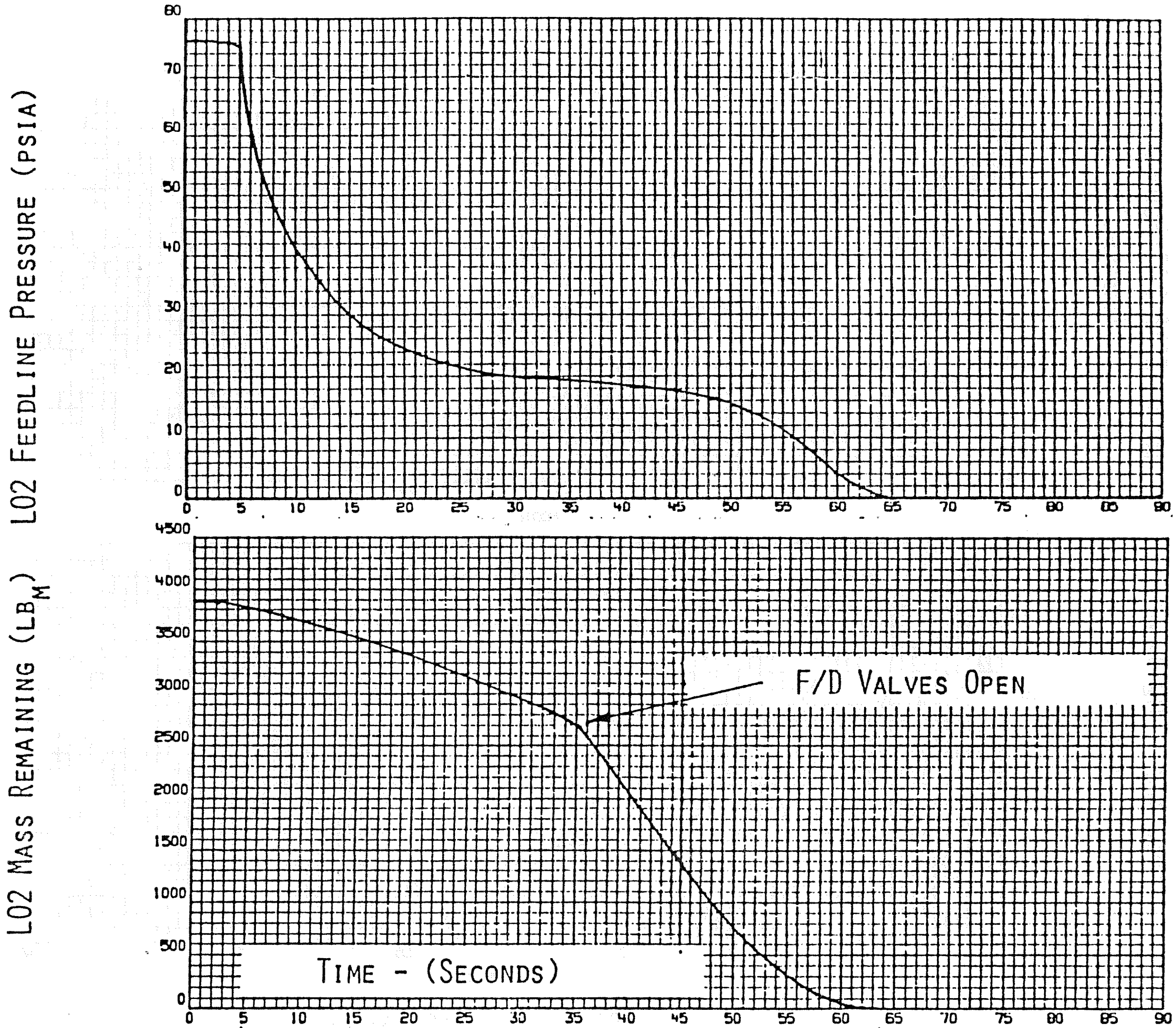


Figure 3.5-31.- Simulation of MPS LO_2 propellant dump/inerting during RTLS abort (MM 602).

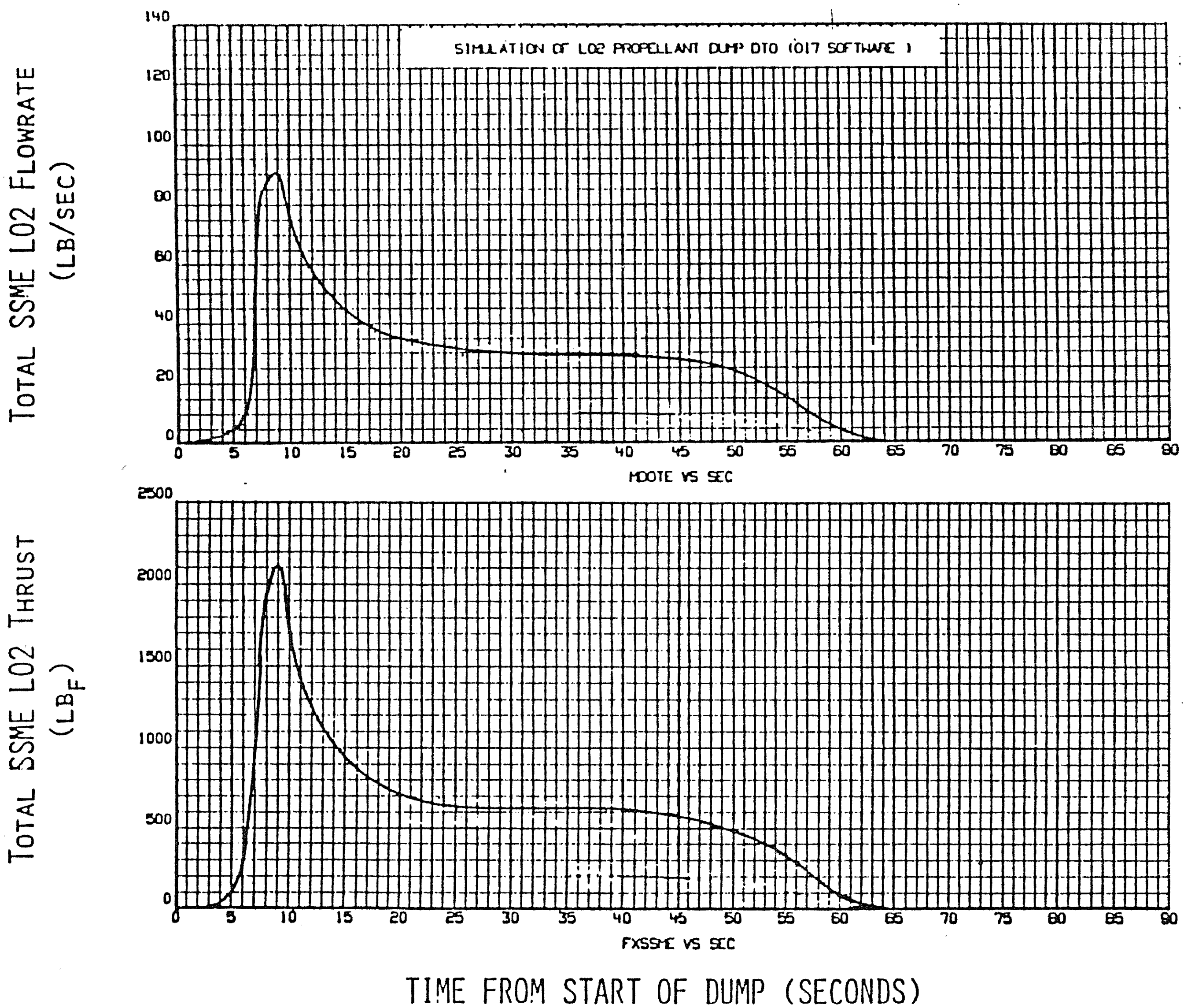


Figure 3.5-32.- LO_2 dump flowrate and force prediction during RTLS abort (MM 602).

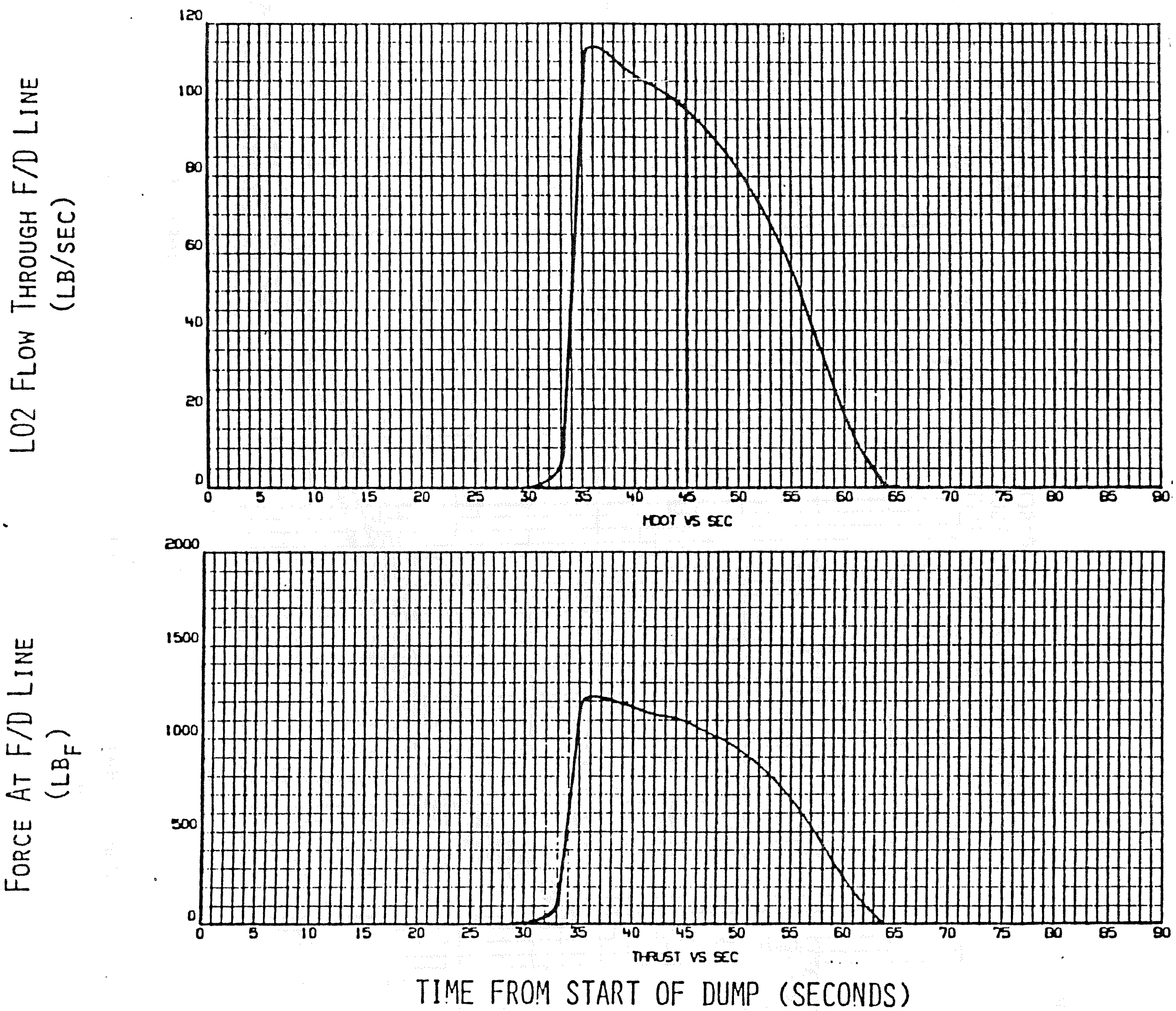


Figure 3.5-33.- LO_2 dump flowrate and force prediction during RTLS abort (MM 602).

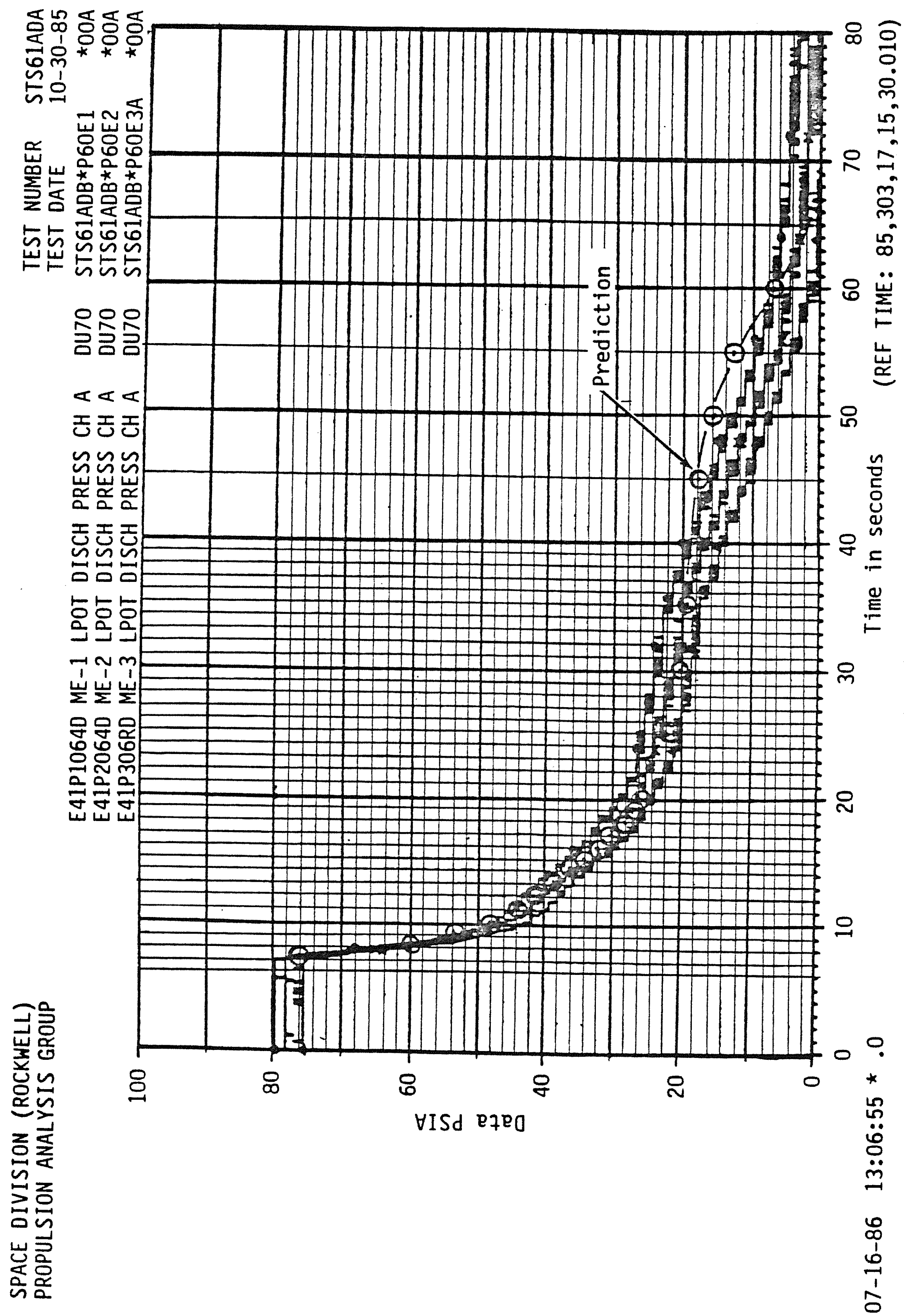


Figure 3.5-34.- Dump model correlation of LO2 feedline pressure during simulated RTLS dump (STS 61A).

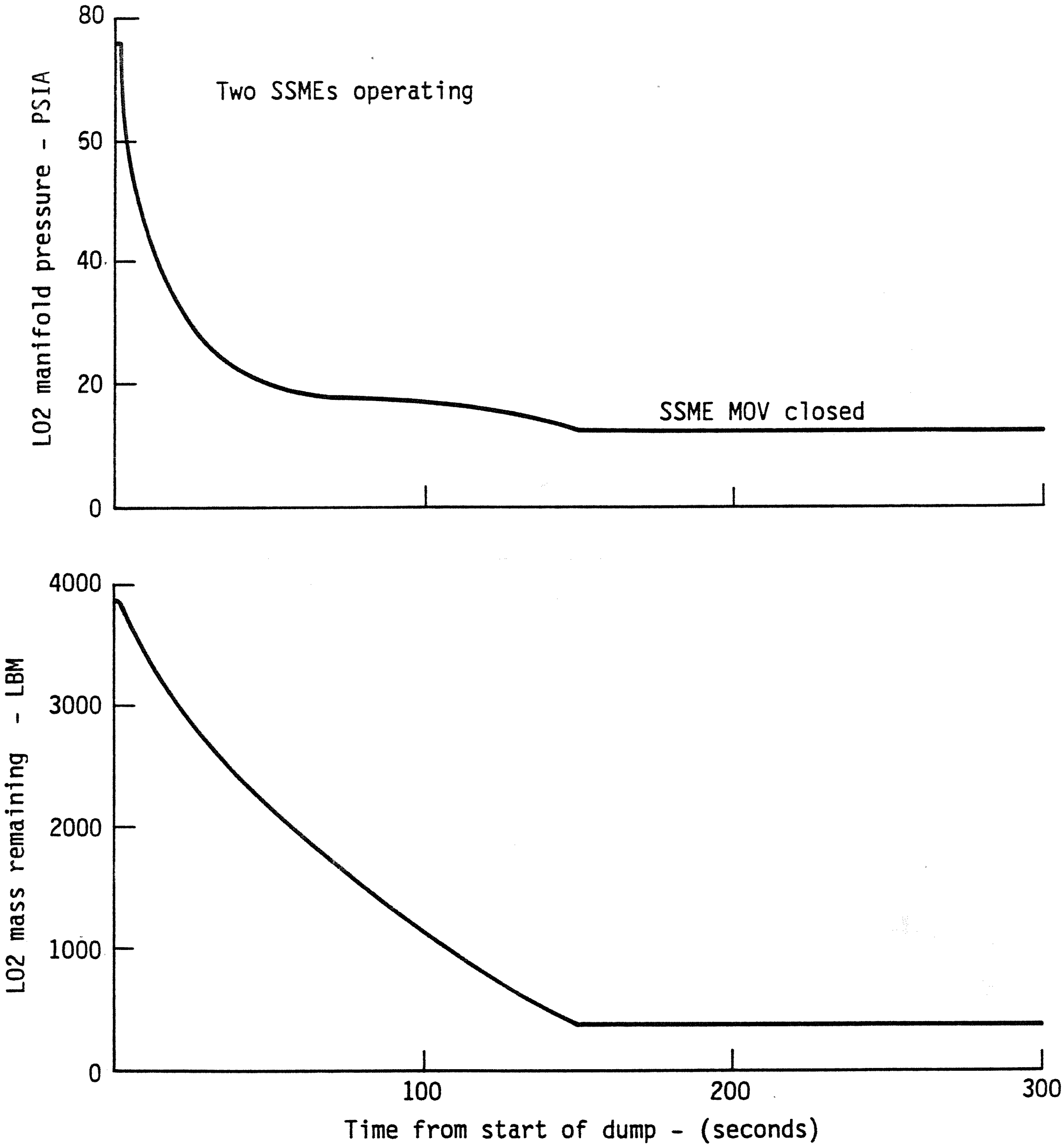


Figure 3.5-35.- MPS LO_2 feed system pressure and mass during RTLS dump (MM 602) (fill/drain valve failed closed).

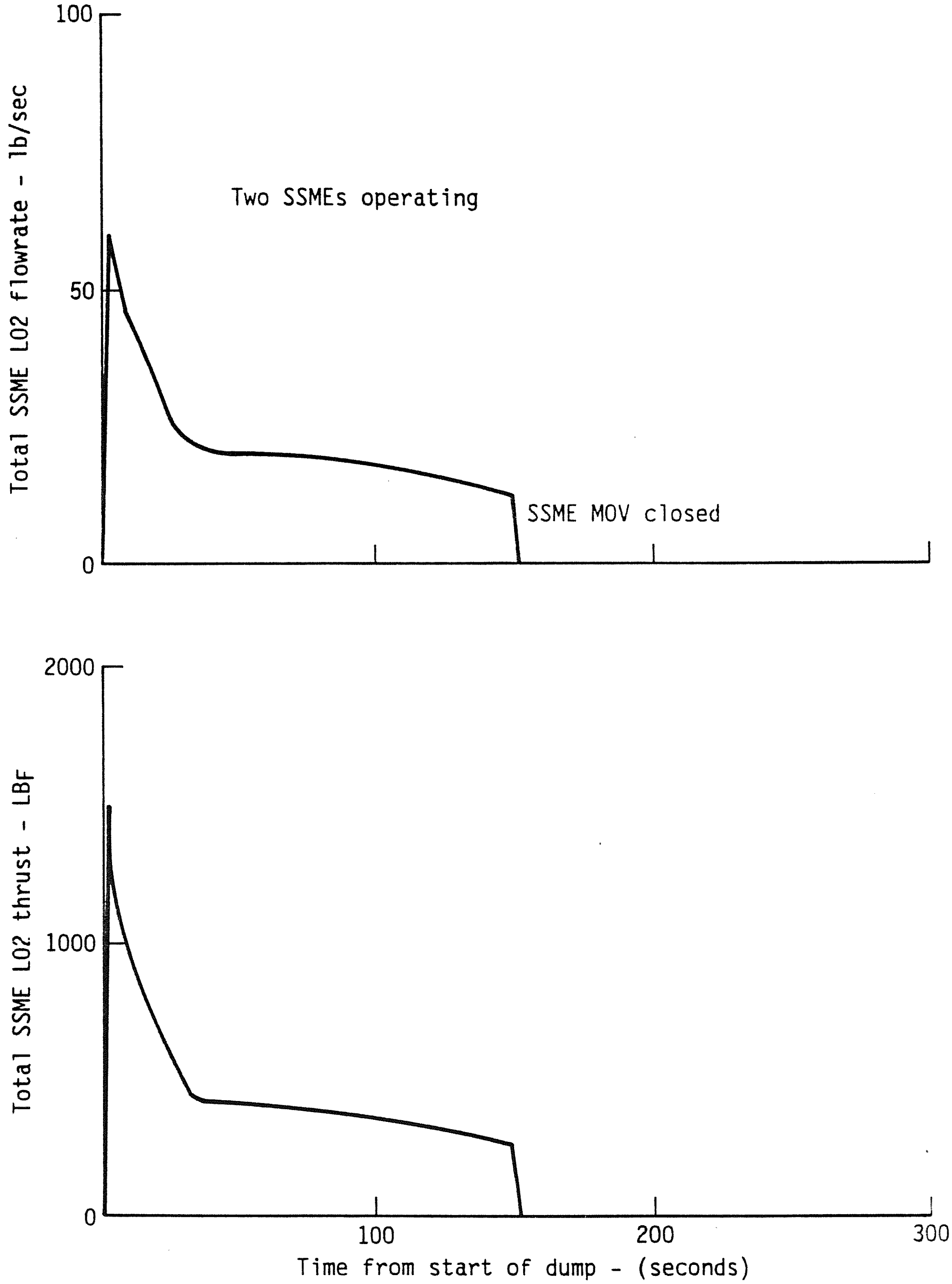


Figure 3.5-36.- MPS LO_2 feed system pressure and mass during RTLS dump (MM 602) (fill/drain valve failed closed).

3.6 MPS VACUUM INERTING

3.6.1 General

MPS vacuum inerting, unpressurized propellant dumping, is the process of opening valves on orbit to expose the MPS to the vacuum environment. The exposure will result in venting overboard any traces of residual propellant remaining after the propellant dump, thereby inerting the system. The subsystems inerted by this process include the LO_2 and LH_2 feed systems, the LH_2 recirculation system, and the GH_2 pressurization system. Vacuum inerting is necessary to do the following:

- Eliminate any potential for pressure buildup in the feedlines which, in the event of a relief valve failure, could overpressure and burst the lines
- Preclude intermittent venting and the associated impacts on vehicle attitude and contamination of experiments
- Preclude any potential hazard from residual propellants during reentry and postlanding

Vacuum inerting is performed after the MPS propellant dump. A second vacuum inerting will be performed if residuals are believed to remain in the manifolds, reference rule 5-67 (ref. 1).

3.6.2 Functional Description

3.6.2.1 Crew Procedure

The crew procedures that define the initiation and termination are contained in the Ascent Checklist Flight Data File and shown in figure 3.6-1. The pneumatic helium isolation valves are verified open to supply helium for the valve operation. Both the LO_2 and LH_2 lines are inerted simultaneously. The LO_2 vacuum inerting is performed by manually opening the LO_2 inboard and outboard fill/drain valves. The LH_2 vacuum inerting is performed by manually opening the inboard and outboard fill/drain valves and the LH_2 topping valve. The LO_2 and LH_2 prevalves are already open since they are left open following the MPS dump. The external tank GH_2 pressurization manifold is vacuum inerted by opening the H_2 pressurization line vent valve for 1 minute.

At the end of the inerting, the crew manually closes the LO_2 and LH_2 outboard fill/drain valves. The LO_2 and LH_2 prevalves and topping valve and the LO_2 and LH_2 inboard fill/drain valves are left open. To conserve electrical power after the completion of the vacuum inerting procedure, the switches for all six prevalves are left in the GPC position. The switches for the LO_2 and LH_2 fill/drain valves and the H_2 pressurization vent line are left in the GND position. With the switches in the GPC and GND positions, power is removed from the opening and closing solenoids of the

P **MPS POWERDOWN (Not AOA)**

R2 MPS ENG PWR (six) - OFF

MPS VACUUM INERTING ACT
5 minutes after MPS dump complete

R2 MPS He ISOL (six) - GPC
✓PNEU He ISOL - OP
✓He I'CNECT (three) - GPC

IF LO2 MAN P < 40
R4 MPS FILL/DRAIN LO2 OUTBD - OP
INBD - OP
OTHERWISE:
✓MCC

R4 MPS FILL/DRAIN LH2 OUTBD - OP
INBD - OP
H2 PRESS LINE VENT - OP
(Start watch)
After 1 minute:
H2 PRESS LINE VENT - GND

NOTE
Expect multiple MAs for
MPS He P as regs bleed down

TIG-5 ▸ P **VACUUM INERTING TERMINATE**

R4 MPS FILL/DRAIN
LH2,LO2 OUTBD (two) - CL
Wait 10 sec .... - GND
LH2,LO2 INBD (two) - GND

R2 MPS PNEU he ISOL - CL

* If MPS PRPLT DUMP B/U LH2 VLV *
* sw was set to OP, set sw to CL *

Figure 3.6-1.- Crew procedure for first vacuum inerting.

corresponding valves. Since the prevalves and the fill/drain valves are pneumatically activated (and not spring-loaded to one position), the valves will remain in their last commanded position. The H_2 pressurization line vent valve is electrically activated; however, it is normally closed (spring-loaded to the closed position). Removing power from the valve solenoid will therefore leave the valve closed.

The final step of the procedure is to close the pneumatic helium isolation valves to isolate the helium supply and to remove power from these normally closed solenoid valves.

3.6.2.2 Vacuum Inerting Duration

Following the MPS dump, vacuum inerting will normally be performed between the OMS-1 burn and the OMS-2 burn. Vacuum inerting should be terminated prior to OMS-2 at TIG-5m (if AOA, TIG-10m) to prevent any contamination from being ingested into the open lines.

The time interval between OMS-1 and OMS-2 is approximately 30 minutes. During the first five flights, vacuum inerting durations varied between 12 minutes and 24 minutes.

3.6.2.3 Vacuum Inerting Failure

The prime failure of concern with vacuum inerting is the failure of the fill/drain valves to open. This failure may produce high LO_2 and LH_2 engine manifold pressures. If these pressures are not relieved by the feedline relief system, rupture of the feedline manifold, further damage to the aft compartment, and possible loss of the vehicle could occur.

The instrumentation for determining the effectiveness of the vacuum inerting procedure includes the LO_2 and LH_2 manifold and engine inlet pressures. If propellant is frozen in the feedlines at the triple point, (triple point pressure of LO_2 is 0.021 psia, and LH_2 is 1.04 psia) following vacuum inerting, the pressures will not be measurable on the instrumentation. Only after melting and vaporization could these quantities be detected and the pressure relieved by a second vacuum inerting.

The details of the crew procedure and MCC recognition of vacuum inerting failures and the required activity in response to these failures are found in the malfunction procedure section of the Booster Console Handbook.

3.6.2.4 On-Orbit Vacuum Inerting

A second vacuum inerting is included in the Orbit Operations Checklist, FDF. If a satisfactory MPS dump and vacuum inerting were performed and the LO_2 or LH_2 engine manifold pressures have not increased, a second vacuum inerting will not be done. Experience from the flights to date has shown that a three-engine dump and the first vacuum inerting inerts the system.

3.6.2.5 Entry Dump, Repress and Purge

At TIG -25 minutes, the center, left, and right helium isolation A valves are open, while the B valves are left closed. Also at TIG -25 minutes the LH_2 and LO_2 outboard fill/drain valves are taken closed by switch throws.

At approximately EI -5 minutes, upon entry to MM 304, the LH_2 manifold is inerted. The LH_2 RTLS dump valves, and inboard fill/drain and topping valves are opened to provide redundant dump paths. The MOV's are commanded open; however, the valves will not respond, since the main engine controllers and EIU's have been powered down. For a TAL case, the controllers and EIU's remain powered up for the entire flight. Nominally the LO_2 inboard fill and drain valves are opened at an inertial velocity of 20K ft/s to inert the LO_2 manifold. At the inertial velocity of 4.5K ft/s (110,000 ft altitude), the LO_2 prevalves and the RTLS dump valves are closed by the GPC's. The manifolds are pressurized with helium through the LO_2 and LH_2 manifold repressurization valves to prevent atmospheric contamination during entry. As a safety measure, the OMS pods, the ET umbilical, and the aft compartments are purged with helium to dilute any possible flammable residuals.

Pressure is maintained in the manifolds on the ground until the engine plugs are installed by the ground crew.

3.6.2.6 References

STS Operational Flight Rules, All Flights, Baseline, JSC-12820, September 1, 1987

3.7 CREW CONTROLS/DISPLAYS

3.7.1 General

The crew can monitor critical MPS parameters using meters and a CRT display. The crew can also use cockpit switches to manually control the positions of some of the valves that are normally controlled automatically by the general purpose computers (GPC's). Manual control of some switches enables the crew to perform certain MPS procedures the GPC's cannot perform due to a failure.

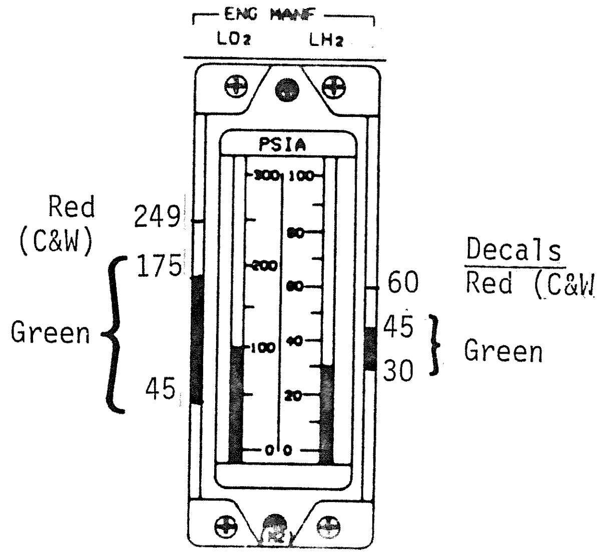
This systems brief will describe the function, nominal usage, and contingency usage of the MPS controls and display.

Component Name: MPS PRESS ENG MANF meters

Panel Location: F7

Component No.: M2

Configuration and Nomenclature:



Function:

Tape meters display the LO_2 and LH_2 engine manifold pressures - that is the pressure between the external tank/Orbiter disconnect valves and the pre-valves.

Nominal Usage:

The meters will be used to monitor the MPS vacuum inert.

Contingency Usage:

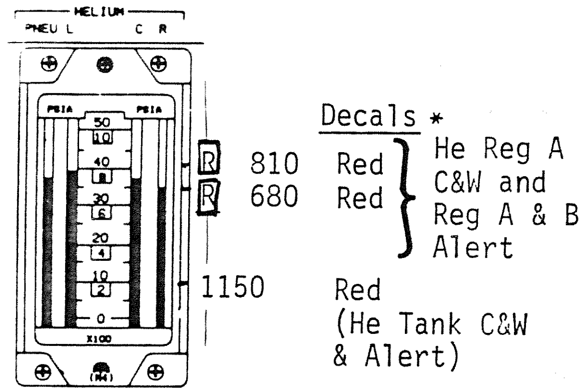
The meters are monitored postburn to allow the crew to take action to prevent over-pressurization of the manifolds in the event of a feedline relief valve or its associated shutoff valve failing closed.

Component Name: MPS PRESS HELIUM meters

Panel Location: F7

Component No.: M4

Configuration and Nomenclature:



Function:

Tape meters display either the four MPS helium supply system pressures or the associated regulator A outlet pressures. Switch S1 (next page) controls which pressures are displayed.

Nominal Usage:

The engine helium meters will be used routinely to monitor engine helium usage during ascent and helium purge flow during entry.

Contingency Usage:

The meters will be used to verify abnormal helium usage on a main engine, to determine the leaking regulator path for isolation purposes and to monitor the system after manual interconnect of the pneumatics tank to that engine. Helium is supplied to each main engine to purge the high pressure oxidizer turbopump intermediate seal. This seal pressure is one of the engine controller internal limit shutdown parameters.

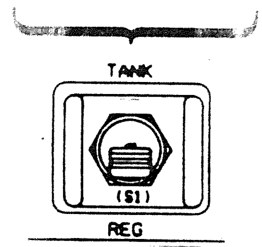
* Note: Alert & Warning is only available for left, center, and right engine helium supplies. Pneumatic supply has no Alert & Warning until OI-8D software implementation.

Component Name: MPS PRESS HELIUM TANK/ REG select switch

Panel Location: F7

Component No.: S1

Configuration and Nomenclature:



Function:

Configures meter M4 for readout of helium tank pressure or helium regulator A outlet pressure sensors.

Nominal Usage:

This switch is nominally left in the TANK position for the purpose of monitoring He tank pressure.

Contingency Usage:

In the REG position, meter M4 will display readings from only the regulator A pressure transducer. The switch is used to determine which sensor source (tank pressure or regulator pressure) has caused the C&W alarm.

Component Name: MPS light (red) C&W panel

Panel Location: F7

Component No.: D7

Note: A class 2 alarm audible tone accompanies this light. Also, the MPS light triggers four MASTER ALARM lights on panel F4, F2, A7A1, and M052J.

Configuration and Nomenclature:

O_2 PRESS	H_2 PRESS	FUEL CELL REAC (R)	FUEL CELL STACK TEMP	FUEL CELL PUMP
CABIN ATM (R)	O_2 HEATER TEMP	MAIN BUS UNDERVOLT (R)	AC VOLTAGE	AC OVERLOAD
FREON LOOP	AV BAY/ CABIN AIR	IMU	FWD RCS (R)	RCS JET
H_2O LOOP	RGA/ACCEL	AIR DATA (R)	LEFT RCS (R)	RIGHT RCS (R)
———	LEFT RHC (R)	RIGHT/AFT RHC (R)	LEFT OMS (R)	RIGHT OMS (R)
PAYLOAD WARNING (R)	GPC	FCS SATURATION (R)	OMS KIT	OMS TVC (R)
PAYLOAD CAUTION	PRIMARY C/W	FCS CHANNEL	MPS (R)	———
BACKUP C/W ALARM (R)	APU TEMP	APU OVERSPEED	APU UNDERSPEED	HYD PRESS

Function:

Light will alert crew to out-of-limit condition on the following parameters:

- MPS ENG LO_2 MANF PRESS - Hi >249 psia
- MPS ENG LH_2 MANF PRESS - Hi >60 psia
- MPS He PRESS C - Lo <1150 psia
- MPS He REG PRESS A C - Hi and Lo >810 psia, <680 psia
- MPS He PRESS L - Lo <1150 psia
- MPS He REG PRESS A L - Hi and Lo >810 psia, <680 psia
- MPS He PRESS R - Lo <1150 psia
- MPS He REG PRESS A R - Hi and Lo >810 psia, <680 psia

Nominal Usage:

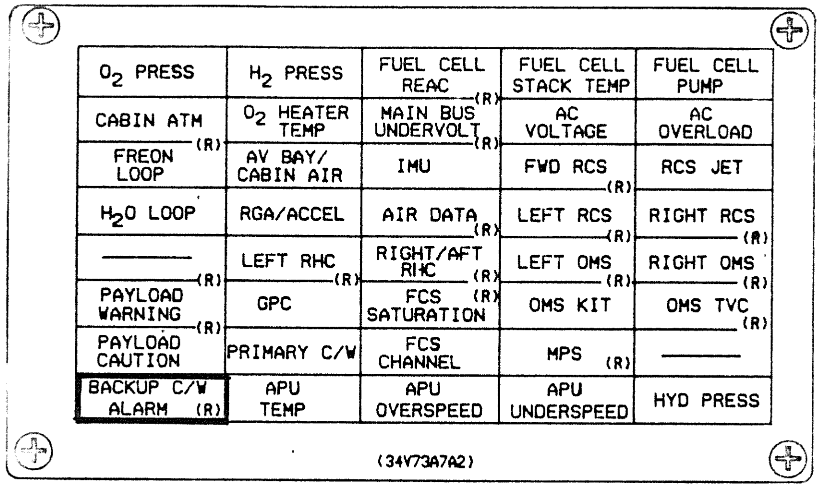
None

Contingency Usage:

MPS ENG LO_2 and LH_2 MANF PRESS high alerts the crew to a high manifold pressure caused by the feedline manifold relief valve or its associated isolation valve failing closed. Crew reaction will be to start the MPS dump, pre MPS dump, or open the fill and drain valves (4) post dump, immediately (panel R2, S1). The helium parameters alert the crew to abnormally low He tank pressure readings or off nominal regulator pressures. Either condition can lead to He starvation for an engine that will cause a redline shutdown. The tank pressure alarm is used in He leak isolation procedures as a cue to interconnect the pneumatics He system to sustain engine operation.

Component Name: Backup C/W Alarm C&W panel

Configuration and Nomenclature:



Panel Location: F7

Component No.:

Note: A class 2 alarm audible tone accompanies this light. Also, the light triggers four MASTER ALARM lights on panel F4, F2, A7A1, and M052J.

Function:

Light will alert crew to out-of-limit condition on the following parameters:

- MPS ENG LO_2 MANF PRESS - Hi $>$249 psia
- MPS ENG LH_2 MANF PRESS - Hi $>$60 psia
- MPS He PRESS C - Lo $<$1150 psia
- MPS He PRESS L - Lo $<$1150 psia
- MPS He PRESS R - Lo $<$1150 psia

Nominal Usage:

None

Contingency Usage:

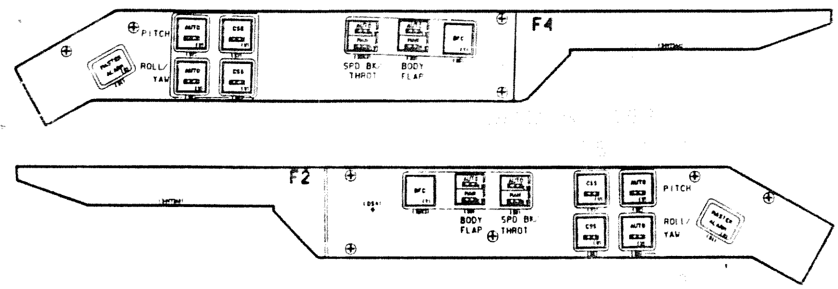
MPS ENG LO_2 and LH_2 MANF PRESS high alerts the crew to a high manifold pressure caused by the feedline manifold relief valve or its associated isolation valve failing closed. Crew reaction will be to start the MPS dump, pre MPS dump, or open the fill and drain valves (4) post dump immediately (panel R2, S1). The helium parameters alert the crew to abnormally low He tank pressure readings that can lead to He starvation for an engine and cause a redline shutdown. The tank pressure alarm is used in He leak isolation procedures as a cue to interconnect the pneumatics He system to sustain engine operation.

Component Name: MASTER ALARM PBI (red light)

Panel Location: F4, F2

Component No: S1

Configuration and Nomenclature



Function:

The red MASTER ALARM light and class 2 alarm audible tone accompany the lights on the C & W panel (location F7). See pages 3.7-5 and 3.7-6.

Nominal Usage:

None

Contingency Usage:

Pushing any MASTER ALARM (MA) PBI turns off the audible tone and MA lights.

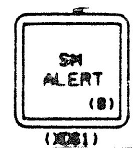
Component Name: SM ALERT (blue light)

Panel Location: F7

Component No.: XDS1

Note: A class 3 alarm audible tone accompanies this light.

Configuration and Nomenclature:



Function:

Light will alert crew to out-of-limit or fault conditions as follows:

- ET LO_2 ULL PRESS - Hi >29 psia and Lo < 0 psia
- ET LH_2 ULL PRESS - Hi >46 psia and Lo <31.6 psia
- MPS He dp/dt L (C,R) - Hi >20 psi/3sec
- MPS He Reg P A L (C,R) - Hi >806 and Lo <679
- MPS He Reg P B L (C,R) - Hi >806 and Lo <679

Nominal Usage:

None

Contingency Usage:

A class 3 alert that is triggered whenever BFS software has determined that a parameter has exceeded the limits as sensed by the BFS GAX program. Crew reaction to the SM alert will be through the applicable cue card procedure. See Malfunctions, section 2 for details of crew and MCC action. For LO_2 and LH_2 ullage pressures, the high limit is used to warn of impending external tank overpressurization, indicating the ET vent/relief valve did not open. The LH_2 low limit alerts the crew that the ET is at the minimum pressure needed to maintain a NPSP of 5.3 psi for the SSME. As a note of explanation, LO_2 ullage pressure is not monitored for a low limit because minimum LO_2 NPSP is not a concern for SSME performance. The inherent weight of LO_2 (16 times greater than LH_2) and the length of the standpipe from the tank to the engines provides sufficient NPSP for the SSME. The MPS helium tank pressure dp/dt alert annunciates excessive helium decay rates possibly created by helium leaks. The SM helium regulator alerts operate the same as in the C/W system, but both A and B regulators are monitored in the SM system.

Display Name: Fault Messages

Configuration and Nomenclature:

See below

Panel Location: N/A

Component No: N/A

Function:

The following fault messages will appear on the Fault message line of the PASS or BFS CRT's for the following causes.

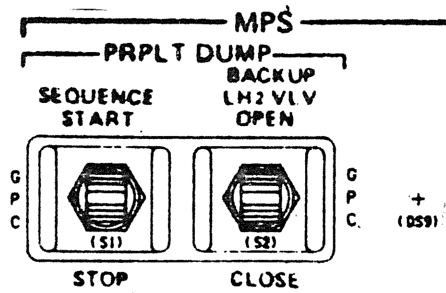
Message	Ops availible		Cause
	PASS	BFS	
ET SEP AUTO	G6		• ET SEP MODE SW FAIL (DEFAULTS TO AUTO)
INH	G1,6	G1, 6	• AUTO SEP INH G1 = BODY RATES & FDLN DISC VLVS G6 = BODY RATES, α AND β
MAN	G1		• ET SEP SW FAIL (RM) - USE SPEC 51
MPS H_e P C(L,R)		G1, 6	• MPS H_e TK P LOW • INDICATED REG P HI/LO • HIGH LEAK RATE dP/dT
MPS LH2/O2 MANF		G1, 6	• MPS LH2 ENG MANIF P HI • MPS LO2 ENG MANIF P HI
ULL		G1, 6	• ET ULL P HI/LO

Component Name: MPS PRPLT DUMP SEQUENCE switches

Panel Location: R2

Component No.: S1, S2

Configuration and Nomenclature:



Function:

The sequence switch selects the mode of starting and stopping the MPS propellant dump sequence. The GPC position allows the computer to automatically initiate the MPS propellant dump sequence at the start of OMS 1. The START and STOP positions provide the capability to manually (through the GPC) initiate and shorten dump intervals for the MPS propellants after MECO confirmed + 20 seconds. The backup switch controls only the operation of the LH_2 RTLS dump valves. The backup switch is not available in the BFS.

Nominal Usage:

These switches will nominally remain in the GPC position.

Contingency Usage:

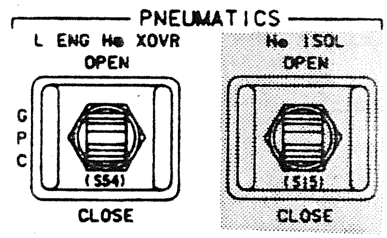
The sequence START position will be used to manually start the propellant dump early. A manually started dump sequence will terminate automatically without positioning the switch to STOP. The STOP position can only be used to shorten the dump time, although there is no defined reason to do this. The start position will also be used to start the MPS dump to relieve manifold pressure buildup (see SCP 2.2.3). The backup switch will be used to dump LH_2 if there is a failure of the LH_2 outboard or inboard fill/drain valve to open during the MPS dump. For GPC/MDM failures pre-MECO, the crew cue card procedures call for the backup switch to be set to open prior to the MPS dump (see SCP 2.2.5).

Component Name: MPS PNEU LEFT ENG He CROSSOVER switch

Panel Location: R2

Component No.: S54

Configuration and Nomenclature:



Function:

This switch controls the solenoid valve downstream of the 4000/750 psi regulators which connects the left main engine helium supply to the MPS pneumatic valve supply. When the switch is in the GPC position, the valve is controlled by the GPC. The OPEN/CLOSE positions override GPC control.

Nominal Usage:

Prior to launch, the switch is left in the GPC position and remains in GPC. The MM 304 automatic dump, inert and repressurization replaces the old manual entry purge prep procedure.

Contingency Usage:

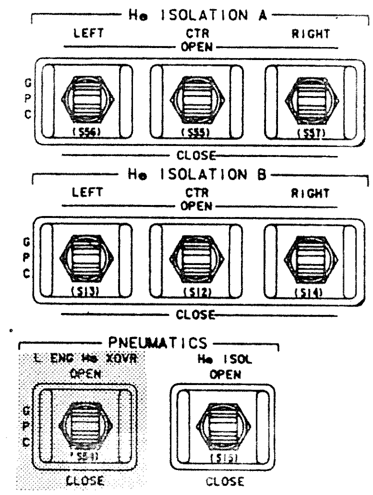
In the event of a lost or leaking open pneumatic helium system, the pneumatic isolation and crossover valves will be opened approximately 30 seconds prior to MECO. This action will provide helium for SSME prevalve closure at MECO.

Component Name: MPS He ISOLATION switches

Panel Location: R2

Component No.: S56, S55, S57, S13 S12, S14, S15

Configuration and Nomenclature:



Function:

These switches control the mode of opening or closing the MPS He isolation solenoid valves downstream of the helium supply bottles for each system. When the switches are in the GPC position, the valves are controlled by the GPC. The OPEN/CLOSE positions override GPC control.

Nominal Usage:

The switches will be in the GPC position until the crew is aboard (approx T-25 min) at which time the switches will be placed to OPEN (the valves will be open for the entire launch). The switches will remain in the OPEN position until the MPS powerdown procedure when the L, C, and R isolation valve A and B switches are set to GPC. The PNEU isolation valve switch is set to GPC during the MPS vacuum inerting termination procedure.

Contingency Usage:

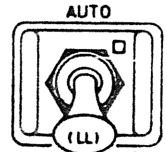
Isolate helium leaks.

Component Name: MPS LH_2 ULLAGE PRESS switch

Panel Location: R2

Component No.: S53

Configuration and Nomenclature:



Function:

This switch controls the mode of opening and closing the GH_2 ullage pressurization flow control valves used for controlling the flow of engine GH_2 to the ET. When in the AUTO position the valves are each controlled independently by an ET pressure transducer. The OPEN position overrides the AUTO position and opens the three valves.

Nominal Usage:

The switch will remain in the AUTO position.

Contingency Usage:

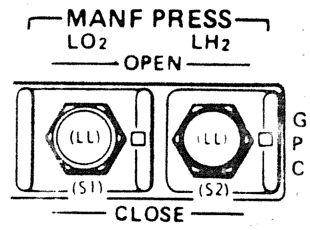
If LH_2 ullage pressure indicates low, the switch will be placed in the OPEN position. The switch will be cycled back to the AUTO position for an ullage pressure reading above 34.5 psia, to prevent venting.

Component Name: MPS MANF PRESS switches

Panel Location: R4

Component No.: S1, S2

Configuration and Nomenclature:



Function:

These switches select the mode of opening and closing solenoid valves which allow helium to flow into the oxidizer and fuel systems. Each switch controls two solenoid valves in series. When the switch is in the GPC position, the solenoid valves are controlled by the GPC. The OPEN/CLOSE positions override GPC control.

Nominal Usage:

These switches will be in the GPC position for launch to allow the GPC to command the solenoids open and closed during LO_2 and LH_2 propellant dump. The MM 304 automatic dump, inert, and repressurization replaces the dd manual entry MPS purge prep procedure. This allows repressurization of the vacuum inerted propellant lines and prevent possible contamination of the lines during entry.

Contingency Usage:

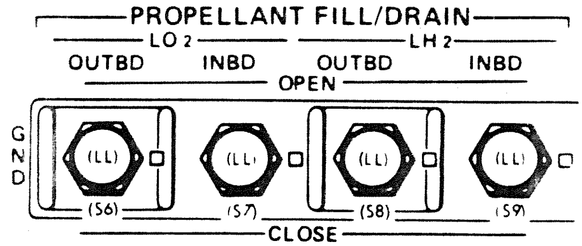
None

Component Name: MPS PROPELLANT FILL/DRAIN switches

Panel Location: R4

Component No.: S6, S7, S8, S9

Configuration and Nomenclature:



Function:

These switches select the mode of opening and closing the LO_2 and LH_2 fill and drain valves. The inboard LH_2 switch also controls the LH_2 topping valve and the high point bleed valve. When the switches are in the GND position, the valves are controlled by the GPC. The OPEN/CLOSE positions override the GPC control.

Nominal Usage:

These switches will be in the GND position from propellant loading through ascent to orbit. During propellant loading the valves will be opened and closed by the LPS via the GPC. The valves will be commanded closed prior to lift-off and will remain closed throughout powered flight. The valves, except for the high point bleed valve, are then opened for the propellant dump and commanded closed approximately 2 minutes later to terminate the dump. After the valves are then closed, the CLOSE commands will be terminated (the valves will remain closed). For the MPS vacuum inerting the LO_2 and LH_2 fill/drain valves will be manually opened (switches to OPEN). At the completion of vacuum inerting, the outboard fill/drain valves will be closed (switches to CLOSE) before all fill/drain valve switches are set to GND. The MM 304 automatic dump, inert, and repressurization replaces the old manual MPS purge prep procedure.

Contingency Usage:

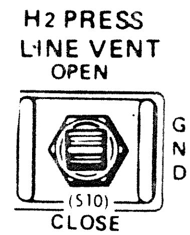
Post dump if the crew is alerted to an overpressurization of the propellant manifolds the respective fill and drain valves will be opened to prevent the manifolds from bursting and overpressing the AFT compartment.

Component Name: MPS H_2 PRESS LINE VENT switch

Panel Location: R4

Component No.: S10

Configuration and Nomenclature:



Function:

This switch selects the mode of opening and closing the ET LH_2 pressurization line vent valve. The valve is controlled by the ground when the switch is in the GND position. The OPEN/CLOSE positions override the computer.

Nominal Usage:

The GND position is used for ground checkout. During the MPS vacuum inerting activation procedure, the switch will be placed in the OPEN position to vacuum inert the ET LH_2 pressurization line. One minute later, the switch is returned to the GND position to close the valve and remove power.

Contingency Usage:

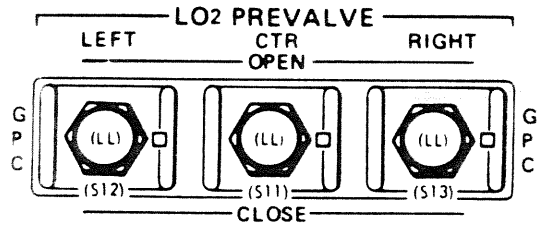
None

Component Name: MPS LO_2 PREVALVE switches

Panel Location: R4

Component No.: S12, S11, S13

Configuration and Nomenclature:



Function:

These switches select the mode of opening and closing the LO_2 prevalves. When the switches are in the GPC position, the prevalves are controlled by the GPC. The OPEN/CLOSE positions override GPC control except during SSME burn when GPC software protection precludes valve closure.

Nominal Usage:

Nominally, the switches remain in the GPC position for prelaunch and ascent. The LO_2 prevalves are commanded open at the start of SSME purge sequence 3 and remain open throughout powered flight with one exception. Should a single SSME malfunction and shutdown prematurely, the prevalve associated with that engine will be commanded closed. At MECO, all prevalves are commanded closed, but are reopened approximately two minutes later for the propellant dump. At the start of entry manifold repressurization the valves are closed.

Contingency Usage:

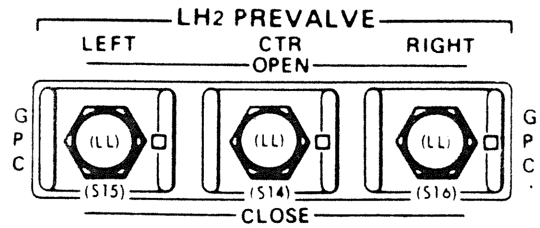
For multiple data path losses (two non-recoverable GPC/FA MDM failures), the three LO2 prevalves will be manually opened prior to the MPS dump.

Component Name: MPS LH_2 PREVALVE switches

Panel Location: R4

Component No.: S15, S14, S16

Configuration and Nomenclature:



Function:

These switches select the mode of opening and closing the LH_2 prevalves. When the switches are in the GPC position, the prevalves are controlled by the GPC. The OPEN/CLOSE positions override GPC control except during SSME burn when GPC software protection precludes valve closure.

Nominal Usage:

Nominally, the switches remain in the GPC position. The prevalves are commanded open at the initiation of propellant loading, closed after the tank is 100 percent full and open 6 seconds before T-0. The valves will remain open until commanded closed following MECO. However, should a single engine malfunction and shutdown prematurely, the prevalve associated with that engine will be commanded closed. The GPC will also command the prevalves open for LH_2 propellant dump. The valves will remain open throughout the continuation of the mission.

Contingency Usage:

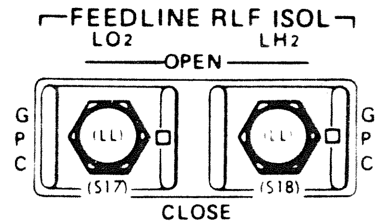
For multiple data path losses (two non-recoverable GPC/FA MDM failures), the three LH2 prevalves will be manually opened prior to the MPS dump.

Component Name: MPS FEEDLINE RLF ISOL switches

Panel Location: R4

Component No.: S17, S18

Configuration and Nomenclature:



Function:

These switches select the mode of opening and closing the LO_2 and LH_2 feedline relief shutoff valves. When the switches are in the GPC position, the valves are controlled by the GPC. The OPEN/CLOSE positions override GPC control.

Nominal Usage:

For all nominal conditions, the switches remain in the GPC position. The computer commands these valves closed prelaunch and they remain closed until MECO when the command is removed and the valve opens.

Contingency Usage:

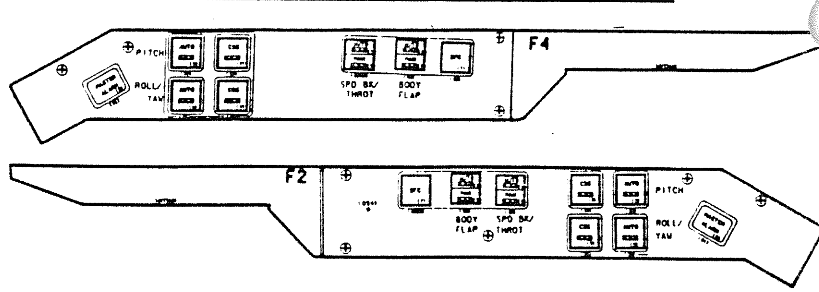
The switches would be used to close the valves in the event of a relief valve failure to reseat. This would be required to maintain pressure in the manifold system during propellant dump and to allow manifold pressurization to prevent contamination during entry. If GPC, MDM, or electrical failures prevent either relief valve from opening at MECO, that switch will be placed to open, allowing the valve to open.

Component Name: BODY FLAP (light)

Panel Location: F4, F2

Component No: S9

Configuration and Nomenclature:



Function:

The BODY FLAP light comes on during the post MECO MPS dump. When the light goes off it indicates completion of the MPS dump.

Nominal Usage:

Indicates completion of MPS dump so that the APU's can be shutdown.

Contingency Usage:

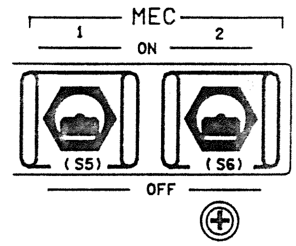
None

Component Name: MEC 1,2 switches

Panel Location: 017

Component No: S5, S6

Configuration and Nomenclature:



Function:

These switches supply redundant power to MEC 1 and 2

Nominal Usage:

The MEC's are powered on prelaunch. After postinsertion, (MET 1:00 hrs) MEC 1 is powered off. Two seconds later MEC is powered off.

Contingency Usage:

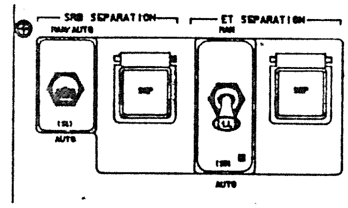
The MEC's are powered off for SRB safing following a Pad Abort.

Component Name: SRB SEPARATION switches

Panel Location: C3

Component No: S1, S2

Configuration and Nomenclature:



Function:

S1 selects the software mode for the SRB separational sequence either MAN/AUTO or AUTO. S2 initiates SRB separation if S1 is in the MAN/AUTO mode.

Nominal Usage:

S1 is set in the AUTO mode.

Contingency Usage:

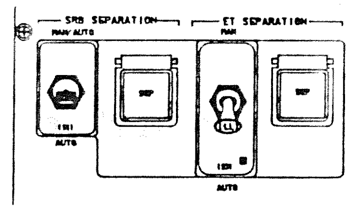
If Roll, Pitch, and Yaw rates, and Dynamic pressure are not within limits for SRB separation, then the crew will get a flashing "SEP INH" message on the pass and BFS Ascent Traj displays. Their procedure is to wait 5 sec for the rates and dynamic pressure to go back down below the limits and for Automatic SRB separation to take place. If, after 5 seconds, separation does not occur, their procedure is to take S1 to the MAN/AUTO position and the push S2 (SRB SEP pb).

Component Name: ET SEPARATION switches

Panel Location: C3

Component No: S3, S4

Configuration and Nomenclature:



Function:

S1 selects the software mode for the ET separation sequence either MAN or AUTO. S2 initiates ET separation if S1 is in the MAN mode.

Nominal Usage:

S1 is set in the AUTO mode. Note: the three main engine red status lights come on at MECO. The lights go out when the ET separates.

Contingency Usage:

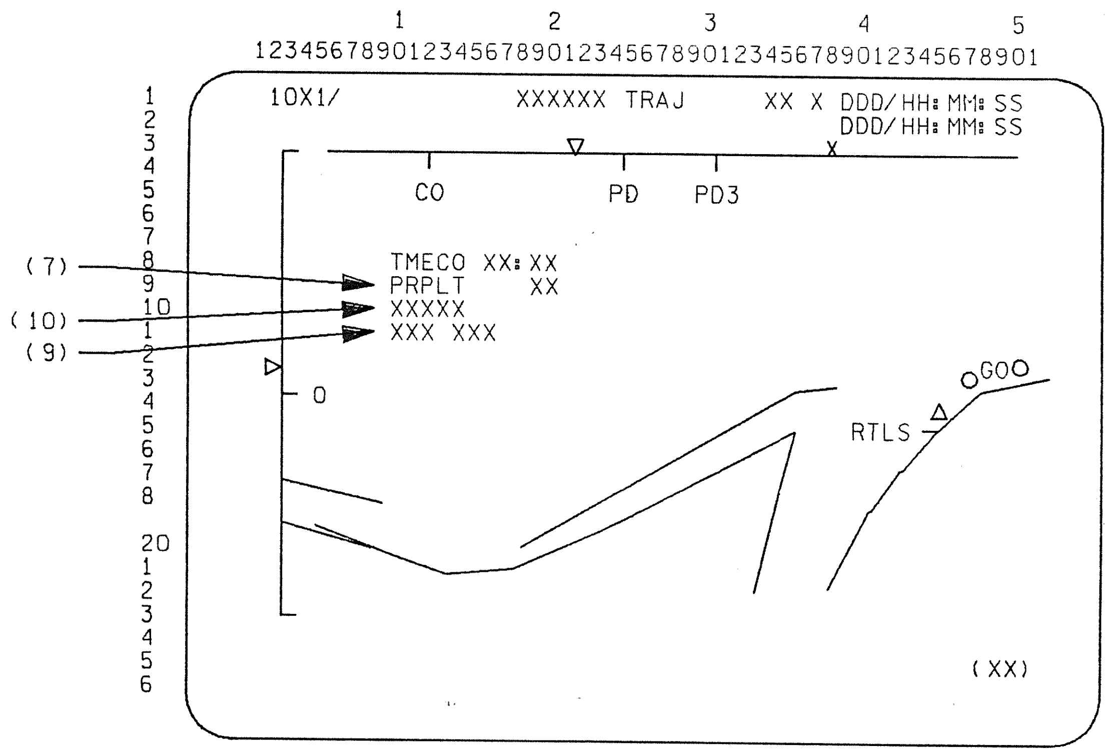
For an "ET SEP INH" fault message caused by failure of the feedline disconnect valves to close post-MECO, S3 is taken to the MAN position. The crew then waits 6 minutes before pushing the ET SEP push button, S4. The switch is taken to the MAN position pre-MECO for an RCS leak. An "ET SEP MAN" message indicates the switch has failed RM, redundancy management, and the crew is required to separate using the SPEC 51 item entry.

Display Name: PASS TRAJ (seen on CRT 1 and 2)

Panel Location: CRT 1 and 2 on panel F7

Component No: N/A

Configuration and Nomenclature:



1904113724.ART;1

Function:

The TRAJ display is the primary PASS display for OPS 101, 102 and 103. It provides the crew with trajectory information following the SSME start command.

Nominal Usage:

"PC<50" appears on the display when the SRB chamber pressure is less than 50 psia (Note(10)). The present propellant remaining in the external tank is also displayed (Note(7)).

Contingency Usage:

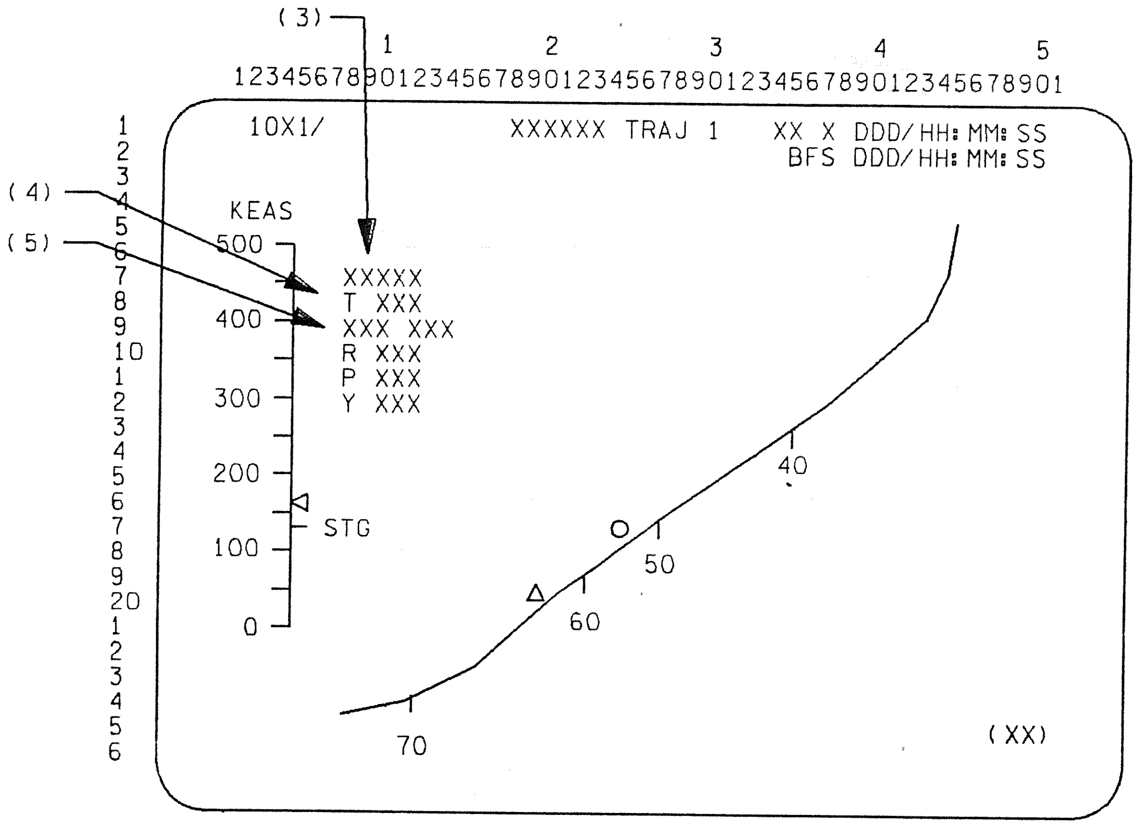
"SEP INH" is displayed flashing on the display if the SRB SEP inhibit discrete is set by the SRB SEP sequence (Note(9)).

<u>Dispay Name:</u> BFS TRAJ1 (seen on CRT 3)

<u>Panel Location:</u> CRT 3 located on F7

<u>Component No:</u> N/A

<u>Configuration and Nomenclature:</u>



190413725.ART, 2

<u>Function</u>:

The TRAJ 1 is the primary BFS display for OPS 101. It is used to monitor guidance performance until SRB staging.

<u>Nominal Usage</u>:

"PC<50" appears on the display when SRB chamber pressure is less than 50 psia (Note (3)). The SSME throttle command is also displayed (Note(4)).

<u>Contingency Usage</u>:

"SEP INH" is displayed flashing on the display if the SRB SEP inhibit discrete is set by the SRB SEP sequence. (Note(5)).

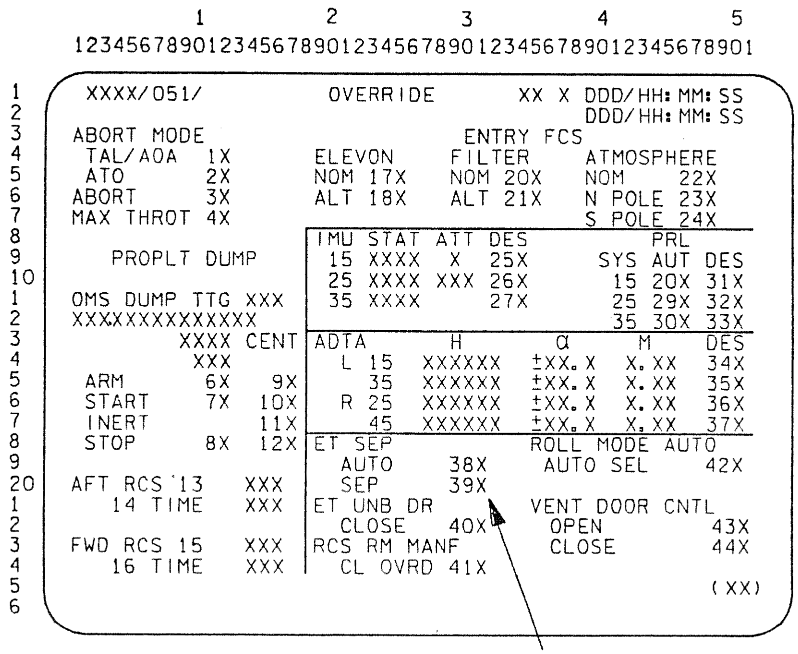
Display Name: OVERIDE (SPEC 51) PASS or BFS

Panel Location: CRT 1,2, or 3 on Panel F7

Component No: N/A

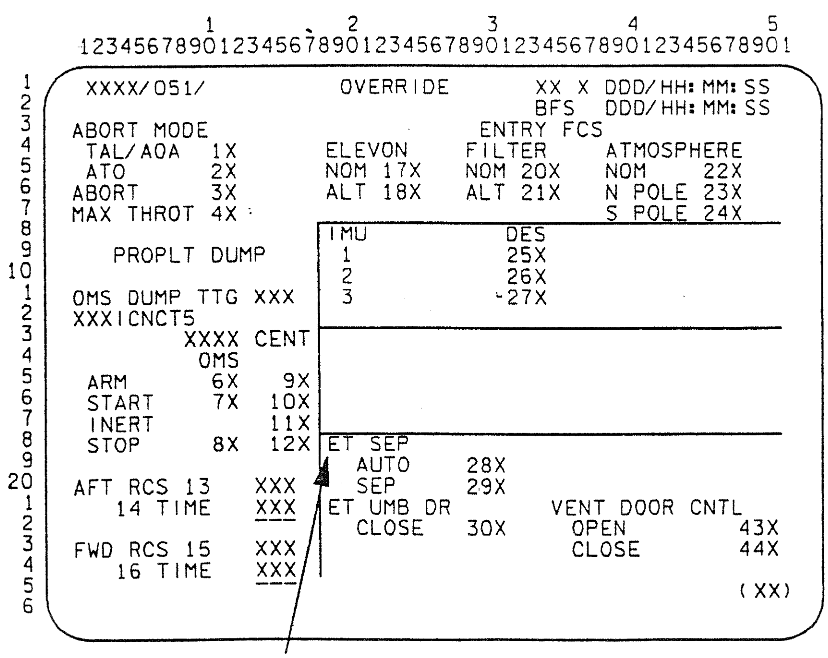
Configuration and Nomenclature:

PASS:



(11) 190413726.ART,2

BFS:



(8)

190413727.ART,1

Function:

OVERIDE GNC SPEC 51 is used during Ops 1,3, and 6. It provides the crew with overide capabilities for the selection of desired switch modes.

Nominal Usage: N/A

Contingency Usage:

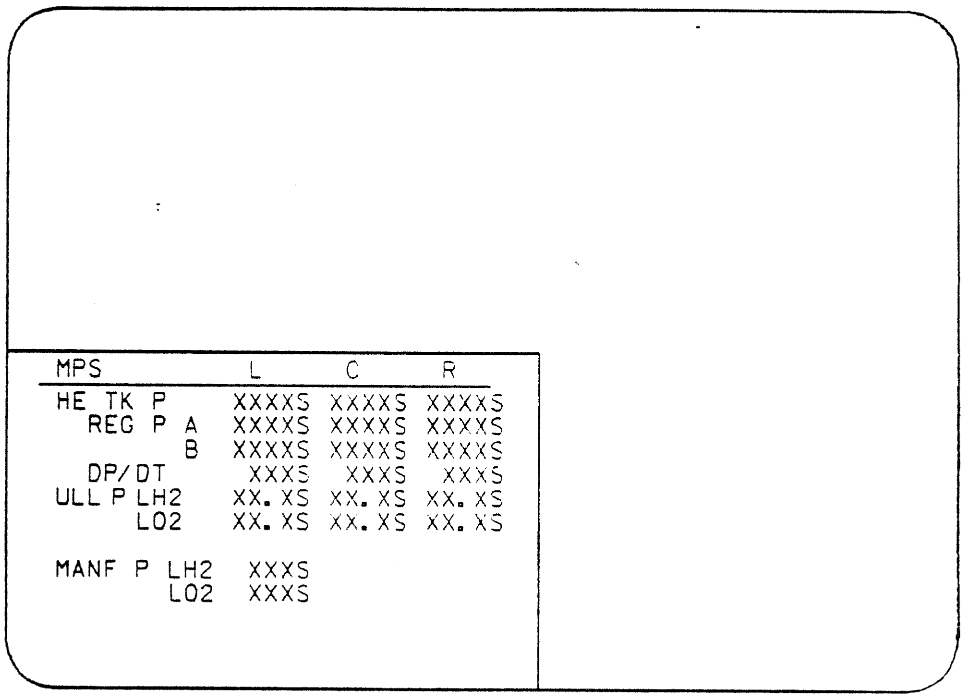
If the redundancy management (RM) on the ET SEPARATION switch detects a contact dilemma, it defaults to the manual mode and displays "ET SEP MAN" fault message and SM alert. The crew can then execute item 38 (item 28 if BFS is engaged) on SPEC 51 to override the default and go back to the AUTO mode for ET SEP. Item 39 (item 29 if BFS is engaged) on SPEC 51 will override "ET SEP INH" fault. This will only be done 6 min after a feedline disconnect valve failure post MECO.

Display Name: GNC SYS SUMM 1 (seen on CRT 1, 2, or 3)

Panel Location: CRT 1, 2, and 3 located on panel F7

Component No.: N/A

Configuration and Nomenclature:



190413728.

Function:

GNC SYS SUMM 1 is a BFS GNC display (DISP 18) available in OPS 1 and 6 via the SYS SUMM key on panel C2. Digital data and/or status information which support aerosurfaces, MPS, Data Processing System (DPS), Flight Control Subsystem Channels (FCS CH), and navigation sensor subsystems are displayed.

Nominal Usage:

Monitor MPS helium system, LO_2 and LH_2 pressurization and MPS manifolds.

Contingency Usage:

Displays digital data and status information with regard to Engine He Tank Pressures (HE TK P), Helium Regulator Pressures A and B (Reg P A and B), dp/dt, LH_2 and LO_2 Ullage Pressure (ULL P), and LH_2 and LO_2 Manifold Pressures (MANF P).

4.1 SRB OVERVIEW

4.1.1 General

The solid rocket booster (SRB) element of the space shuttle (fig. 4.1-1) is made up of six subsystems:

- The solid rocket motor
- The structural subsystem
- The thrust vector control subsystem
- The separation subsystem containing mechanical and ordnance equipment
- The recovery subsystem containing mechanical and parachute equipment
- The electrical subsystem including the range safety system (RSS)

4.1.2 Solid Rocket Motor

The solid rocket motor (SRM), shown in figure 4.1-2, is the primary propulsive element providing impulse and thrust vector control (TVC) from ignition to SRB staging. The SRM consists of a lined, insulated, segmented rocket motor case loaded with solid propellant; an ignition system complete with electromechanical safe and arm device, initiators, and loaded igniter; a movable nozzle; systems tunnel bracketry; instrumentation; and the necessary integration hardware.

The major configurations required to assemble one SRM are a forward rocket motor segment, two center rocket motor segments, an aft rocket motor segment, an aft exit cone assembly, and a nozzle ordnance ring. The SRM components and subsystems are physically interchangeable and replaceable. Performance, interchangeability, and replaceability between a flight set of SRM's can be maintained by matching the burning rates of motor segments cast in matched pairs from the same propellant lot. Sea level thrust for an SRM will be 3.3 million lb. The two SRM's provide 71.4 percent of the thrust at liftoff and during first stage. SRM separation occurs at an approximate altitude of 150,000 ft.

4.1.2.1 Case

The function of the case is to provide a pressure vessel in which thrust can be developed. Each case consists of 11 segments. The material used in the construction of each case segment is D6AC steel. The details of SRB casing are discussed in systems brief 4.2.

4.1.2.2 Propellant

The propellant is a composite-type solid propellant. The propellant grain is designed to fulfill the performance requirements of the shuttle program. Systems brief 4.3 includes descriptions of the propellant formulation, grain

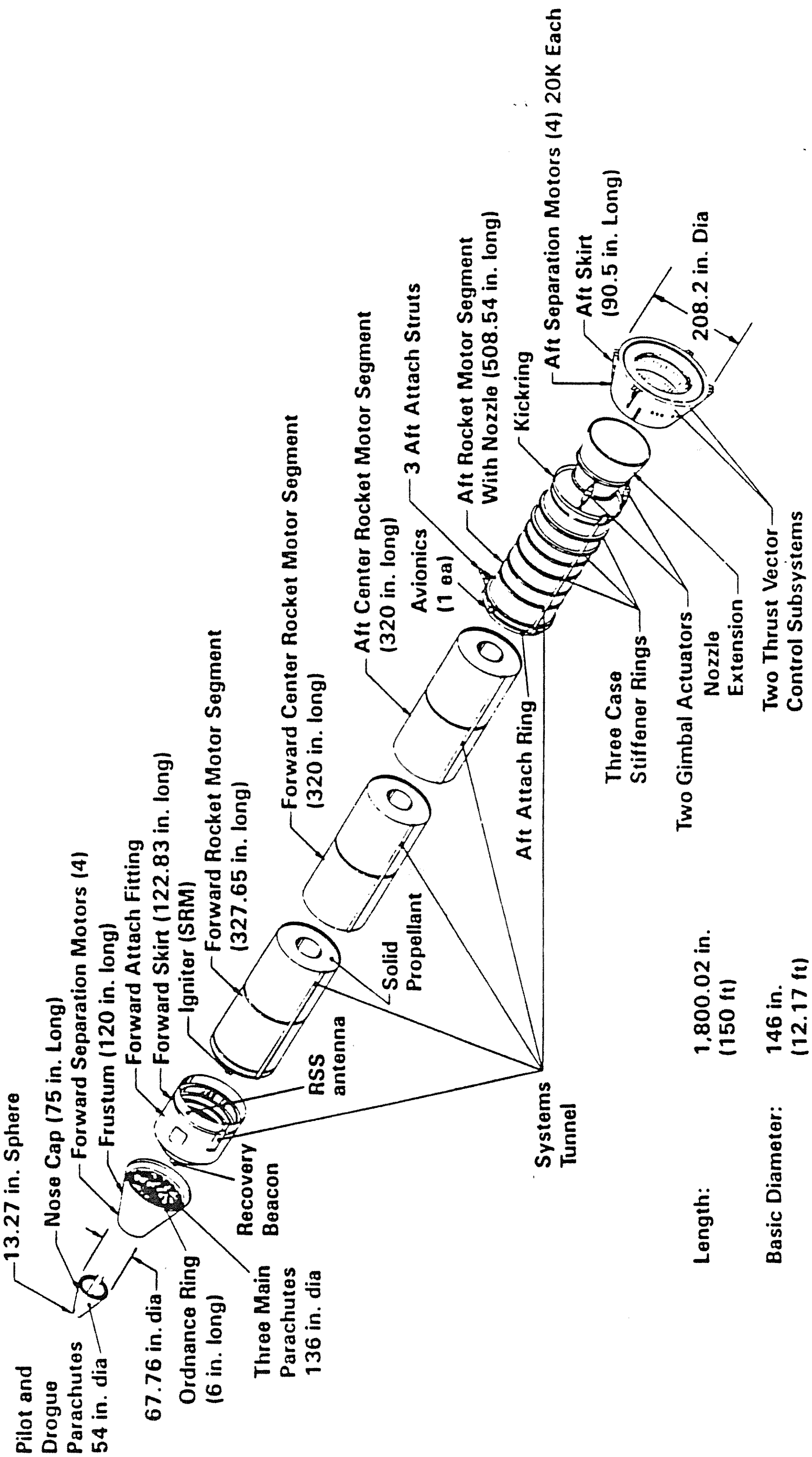


Figure 4.1-1.- SRB components location.

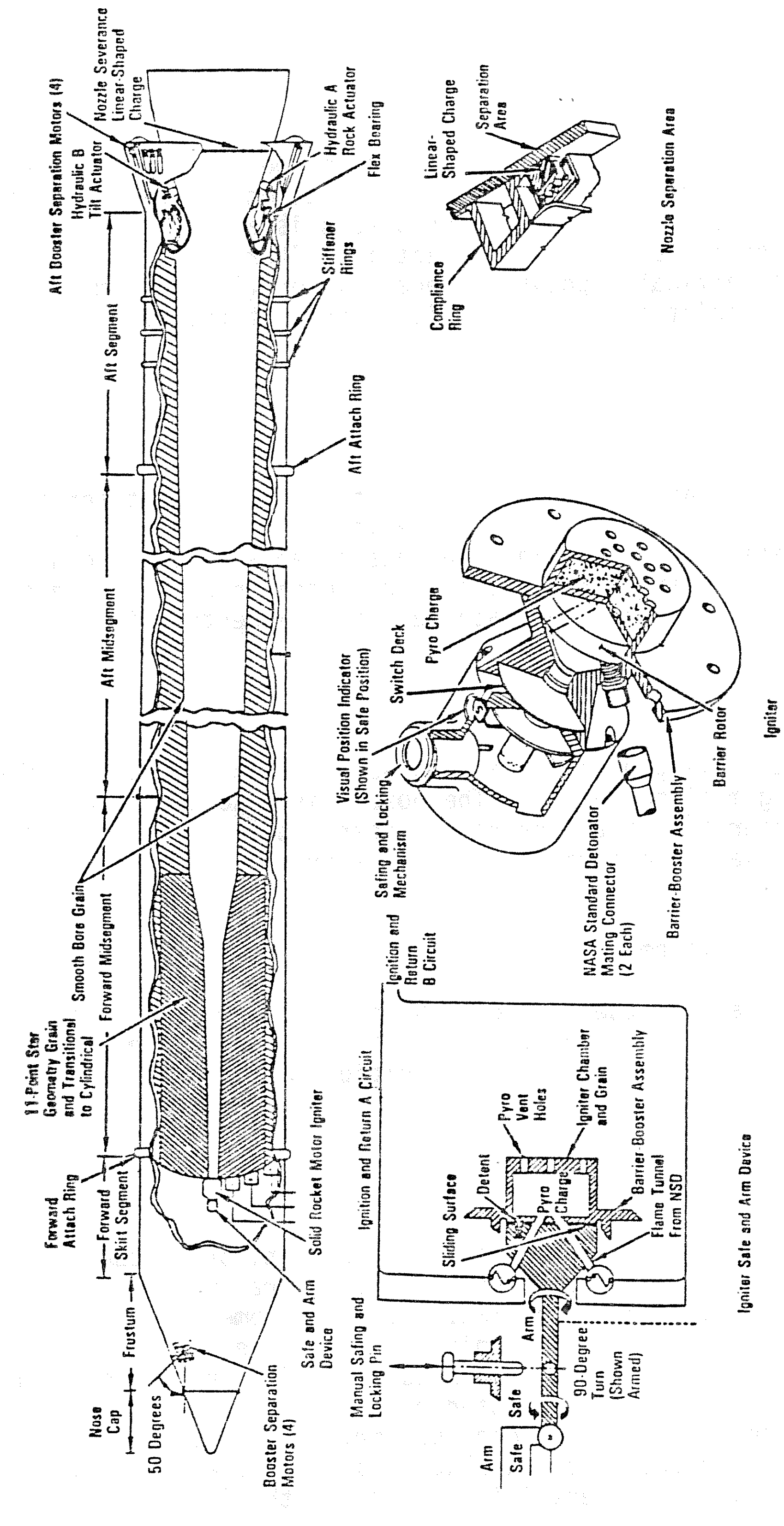


Figure 4.1-2.- SRB solid rocket motor.

design, burn rate, manufacturing process, propellant mixing, and propellant casting.

4.1.2.3 Insulation

The insulation includes chamber insulation, propellant relief flaps, and forward and aft facing inhibitors. The insulation material used for the chamber, relief flaps, and forward inhibitors is an asbestos-silica-filled nitrile butadiene rubber (NBR). The aft inhibitor material is an asbestos-filled carboxyl terminated polybutadiene polymer (CTPB polymer). The purpose of the insulation in the SRM is to protect the cases during motor operation.

4.1.2.4 Liner

The liner material has been designed to provide a bond between the insulation and propellant in SRM's. The liner is compatible with the selected insulation and propellant and has acceptable aging characteristics to maintain the insulation/liner/propellant bond throughout the service life of the SRM. The selected liner material is an asbestos-filled CTPB polymer.

4.1.2.5 Nozzle

The nozzle is a convergent-divergent movable design that uses a flexible bearing as the gimbal mechanism. The nozzle is also constructed to be able to withstand water impact and retrieval operations. The assemblies that make up the nozzle are described in systems brief 4.4.

4.1.2.6 Ignition System

The ignition system for the SRM consists of a forward end internally-mounted solid-rocket-type igniter, an igniter initiator, an igniter adapter, and a safe and arm device. Details of the ignition system are found in systems brief 4.5.

4.1.2.7 Instrumentation

The instrumentation supplied with the SRM consists of pressure transducers. For operational flight instrumentation (OFI), three motor chamber pressure transducers are installed on the SRM. For development flight instrumentation (DFI), one main igniter chamber pressure transducer is installed in addition to the OFI transducers. The triple-redundant motor chamber pressure transducers are used primarily by the orbiter for SRB separation and secondarily for motor performance monitoring. The single main igniter chamber pressure transducer is used to evaluate ignition system performance during development flight tests of the SRM.

4.1.2.8 Manufacturing Operations

Each steel segment is manufactured by hot forging and cold forming operations from a 31,000-lb D6AC steel billet. After the segments are formed and rough machined, they are heat treated, finish machined, inspected, and proof tested. The segments are then shipped to the Thiokol facility in Utah for loading, where they are solvent cleaned in preparation for assembly and loading.

4.1.3 Structures

The SRB structural subsystem, shown in figure 4.1-3, provides the necessary structural support for the shuttle vehicle on the launch pad; transfers thrust loads to the orbiter and ET; and provides the housing, structural support, and bracketry needed for the recovery system, the electrical components, the separation motors, and the TVC system. This subsystem consists of the nose cone assembly (frustum and nose cap), the forward ordnance ring, the forward skirt including the forward SRB/ET attach fitting, the aft SRB/ET attach ring and attach struts, the aft skirt including the heat shield, and the systems tunnel and structure of mounting other SRB subsystems components. In addition, frustum flotation, weighing, hoisting and towing provisions, and structural thermal protection are provided. Most structures components will be used for 10 to 20 flights.

4.1.3.1 Nose Cone Assembly

The nose cone assembly, shown in figure 4.1-4, includes the nose cap, the frustum structure, the frustum flotation components, and hardware for attaching the forward separation motors and nose cap thrusters. The nose cap is basically an aluminum monocoque structure with a hemispherical section at the forward end. The base is 68 inches in diameter, and the overall length is 75 inches. This structure is a riveted assembly consisting of machined 2024 aluminum sheet skins, formed ring segments, machined fittings, formed cap, and a machined separation ring, and weighs about 290 lb. The nose cap houses both the pilot and drogue parachutes and is separated from the frustum by three nose cap thrusters. They are fired at approximately 15,704 feet altitude during the entry phase. The nose cap is initially positioned/oriented on the frustum by six alignment pins, which serve as guides until sufficient clearance has been attained between the cap and frustum during separation.

The frustum structural assembly is composed of machined 2219 aluminum shear beams, ring, fittings, separation motor supports, main chute supports and 7075 aluminum formed skins, and weighs about 3200 lb. The frustum minor base diameter is 68 inches, the height is approximately 10 feet, and the major base is 146 inches in diameter. The frustum houses the main parachutes, provides the structural support for the forward separation motors, and incorporates flotation devices and handling hardware for water recovery.

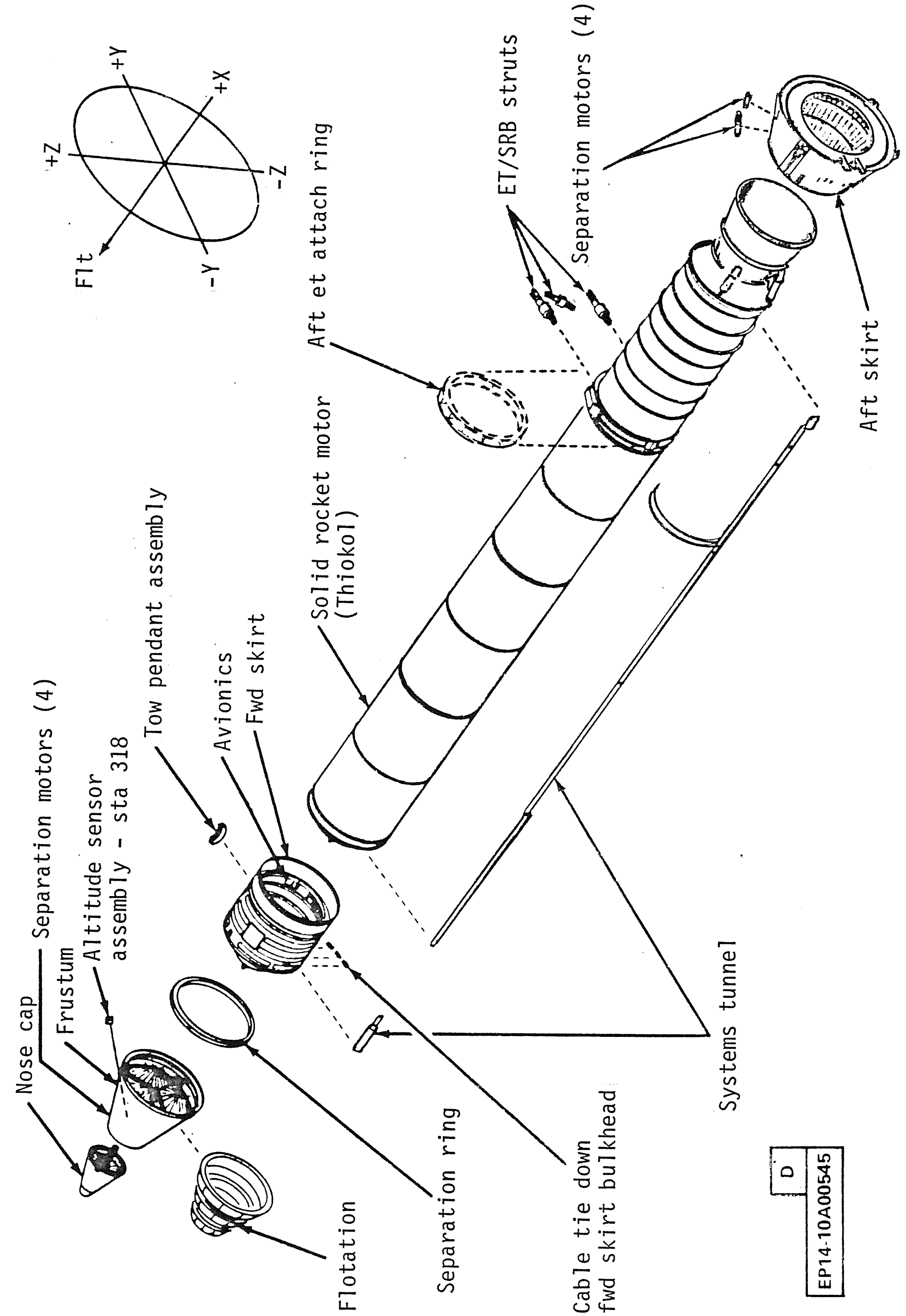


Figure 4.1-3.- SRB structural subsystem.

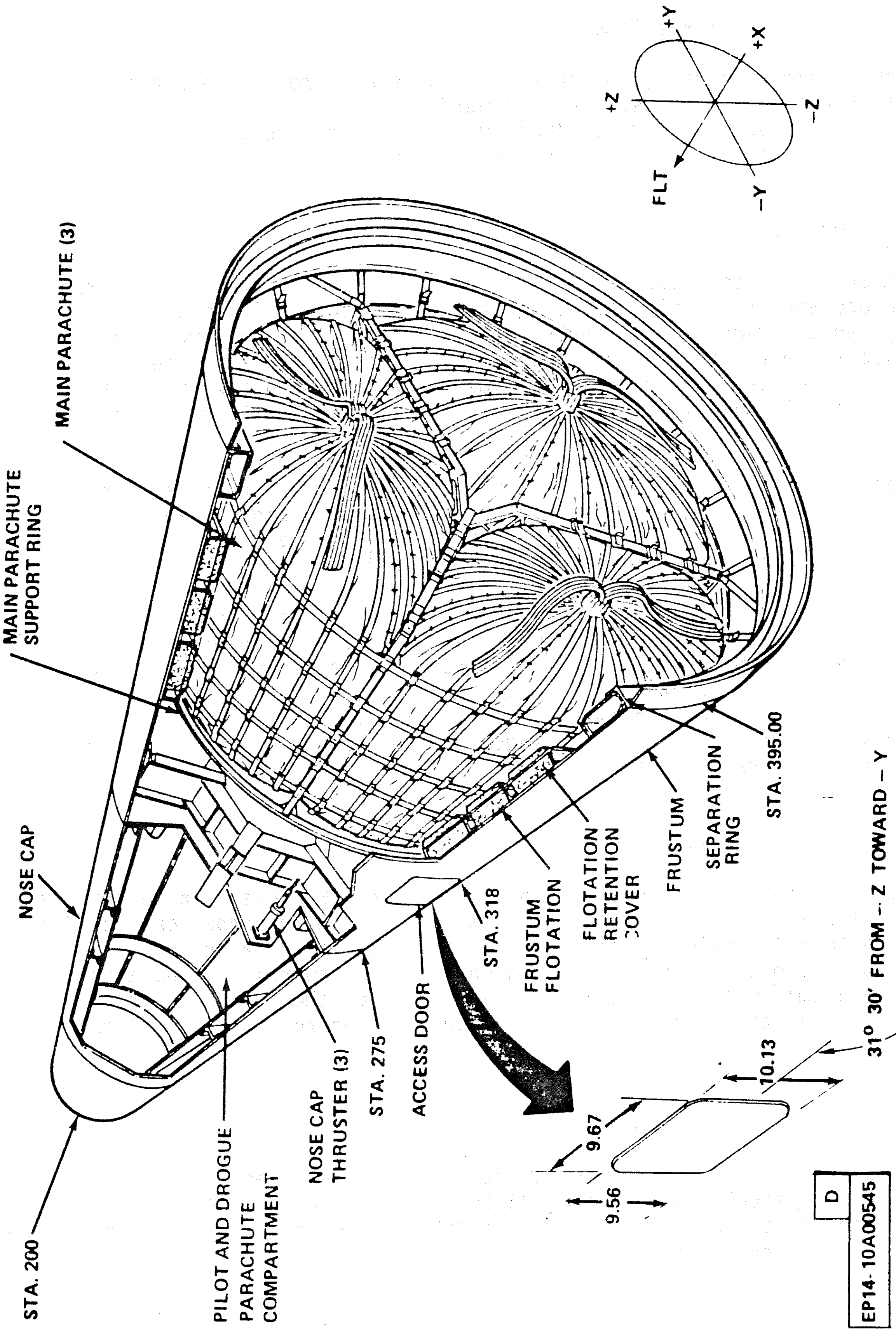


Figure 4.1-4.- SRB nose cap and frustum.

4.1.3.2 Forward Ordnance Ring

The forward ordnance ring, 146 inches in diameter, provides a plane of separation between the frustum and forward skirt assemblies. The ring is precision-machined from a 2219 aluminum ring forging and provides mounting provisions for the liner-shaped charge used in the severance function.

4.1.3.3 Forward Skirt

The forward skirt comprises all structure between the forward SRM segment and the ordnance ring (fig. 4.1-5). It includes an SRB/ET attachment fitting, which transfers the thrust loads to the ET, and a forward bulkhead which seals the forward end of the skirt. The skirt provides the structure to react parachute loads during deployment, descent, and towing. The tow pendant assembly mainly comprises formed sheet metal parts, nylon and steel cables.

Secondary structure is provided for mounting components of the electrical and instrumentation (E&I) subsystem, the rate gyro assembly, range safety panels, and the systems tunnel components. The skirt assembly at completion of installation is sealed to provide additional flotation capability.

The forward skirt, 146 inches in diameter and 125 inches long, consists of a 2219 aluminum welded cylinder assembly made from precision-machined and brake-formed skin panels and a welded thrust post subassembly weighing approximately 5700 lb. Welded to this are the forward ring, SRM attach ring, intermediate rings, gussets, beams, etc. The skirt is then artificially aged and machine finished. Master controlled hole patterns are drilled for component mounting.

4.1.3.4 Systems Tunnel

The SRB systems tunnel, located outboard of each SRB, houses the electrical cables associated with the E&I subsystem and the linear-shaped charge of the RSS. The tunnel provides lightning, thermal, and aerodynamic protection and mechanical support for the cables and charge. The tunnel is manufactured from 2219 aluminum alloy material and extends from the forward skirt along the motor cases to the aft skirt. The tunnel is approximately 10 inches wide and 5 inches high.

4.1.3.5 SRB/ET Aft Ring and Attachments

The SRB/ET aft ring and attachments are composed of a steel ring and three struts that physically attach the SRB to the ET. These attachments are designed to react loads in place of the attach ring and allow unrestrained contraction/expansion of SRB or ET in the longitudinal direction.

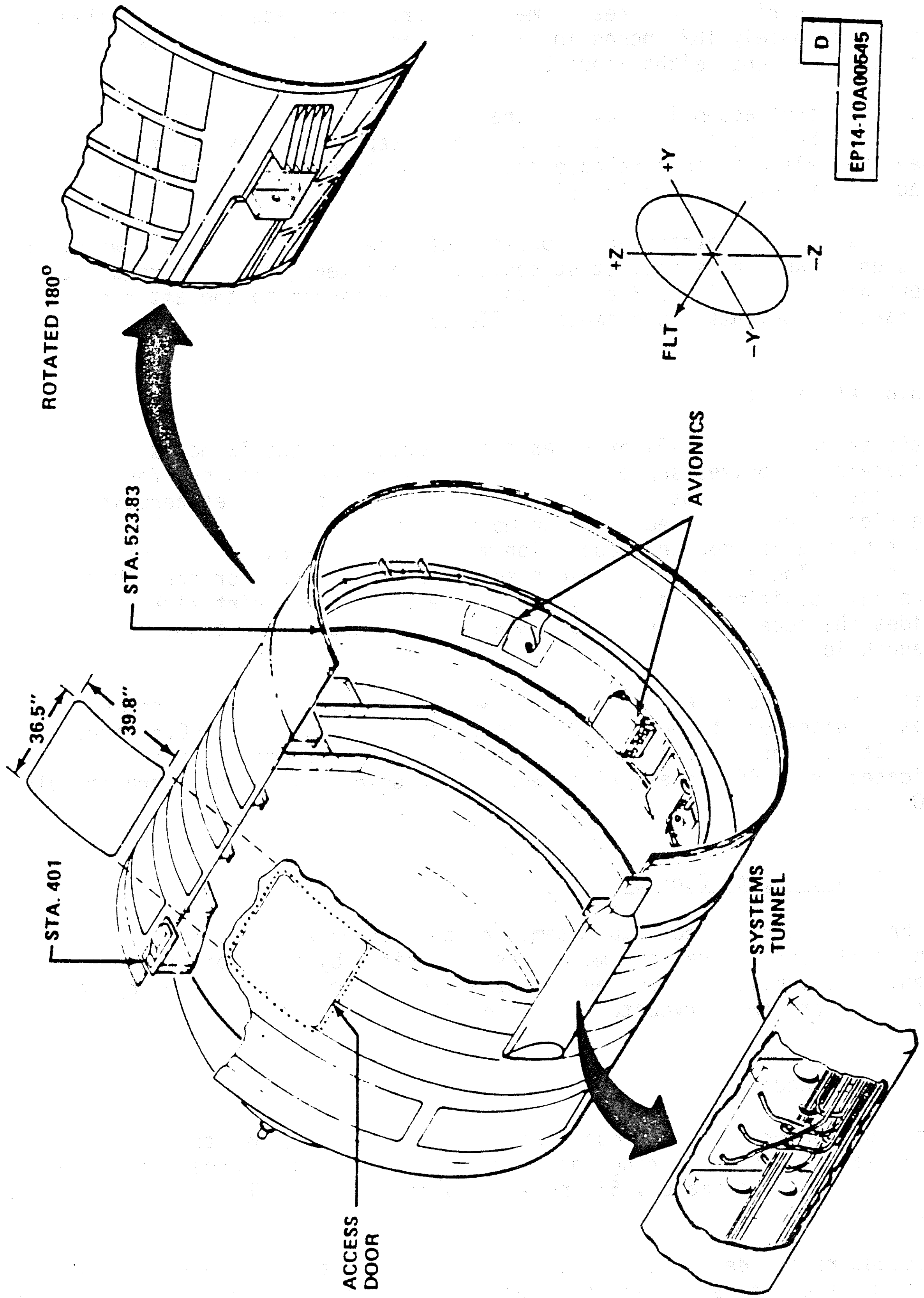


Figure 4.1-5.- SRB forward skirt assembly.

The aft attach ring is a three-segment construction, made from 4340 steel; it is approximately 162 inches in diameter, an average of 7 inches high, and 16 inches wide, and weighs about 3700 lb.

The attach strut assembly provides the physical attachment between the aft ring of the SRB and the ET. It incorporates separation devices midway between the elements that release the SRB upon command. The upper portion of each strut remains with the ET.

A set of struts and attachments consists of three each of the following: ET clevis end, SRB clevis end, strut segment, adjustment butt, and required washers and bolts. The SRB end of the struts attaches to the aft ring. This hardware weights approximately 1550 lb.

4.1.3.6 Aft Skirt

The aft skirt (fig. 4.1-6) provides attach points to the launch support structure and provides support to the shuttle on the launch pad for all conditions prior to booster ignition. The aft skirt provides aerodynamic protection, thermal protection, and mounting provisions for the TVC subsystem, the aft mounted separation motors, and the E&I subsystem components. The aft skirt provides sufficient clearance for the SRM nozzle at the null position plus full gimbal travel. The aft skirt kick ring provides the necessary structural capability to absorb and transfer induced prelaunch loads.

The aft skirt structure assembly is a welded and bolted conical shape, 146 inches in diameter at the top, 208 inches at the bottom, and 90.5 inches long. It is configurated for left- and right-hand assemblies and is fabricated using D6AC steel rings and 2219 aluminum, weighing approximately 11,700 lb.

4.1.4 Thrust Vector Control

The thrust vector control subsystem, in conjunction with the SRM, provides pitch, roll, and yaw vehicle movements as desired by the orbiter command system. The subsystem, mounted in the aft skirt, consists of two hydraulic power units and two servoactuators (fig. 4.1-7).

4.1.4.1 Servoactuators

The servoactuators, linear double acting type, are attached to the aft skirt just forward of the kick ring and to the SRM nozzle compliance ring. The actuators are approximately 53 inches long pin to pin. The actuator stroke is ±6.4 inches.

The actuators are designed to retain the nozzle in the null position throughout the separation and to water impact. As shown in figure 4.1-8 the actuators are oriented at 45°, outboard, to the vehicle pitch and yaw axis.

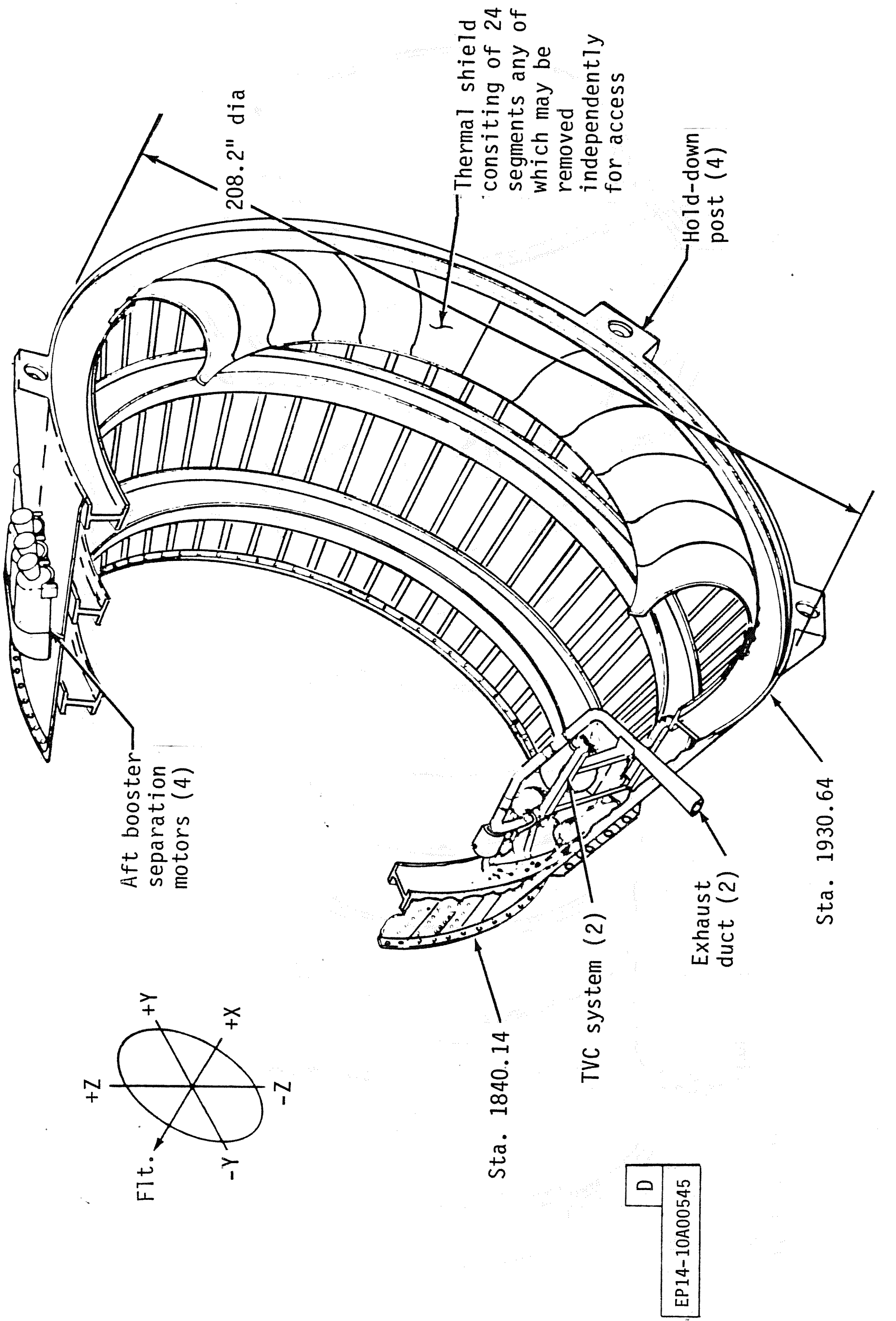


Figure 4.1-6.- SRB aft skirt.

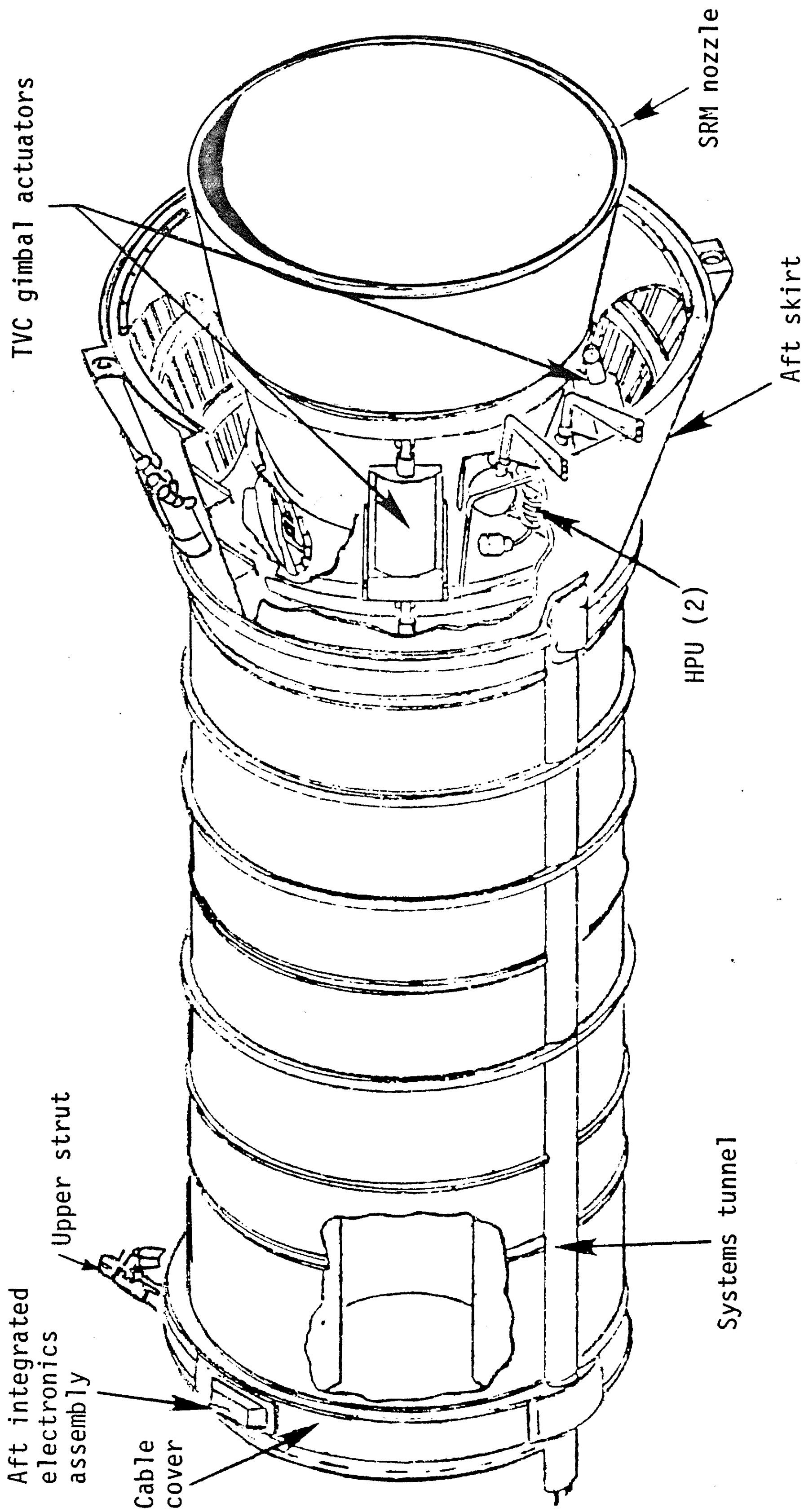


Figure 4.1-7.- SRB TVC components location.

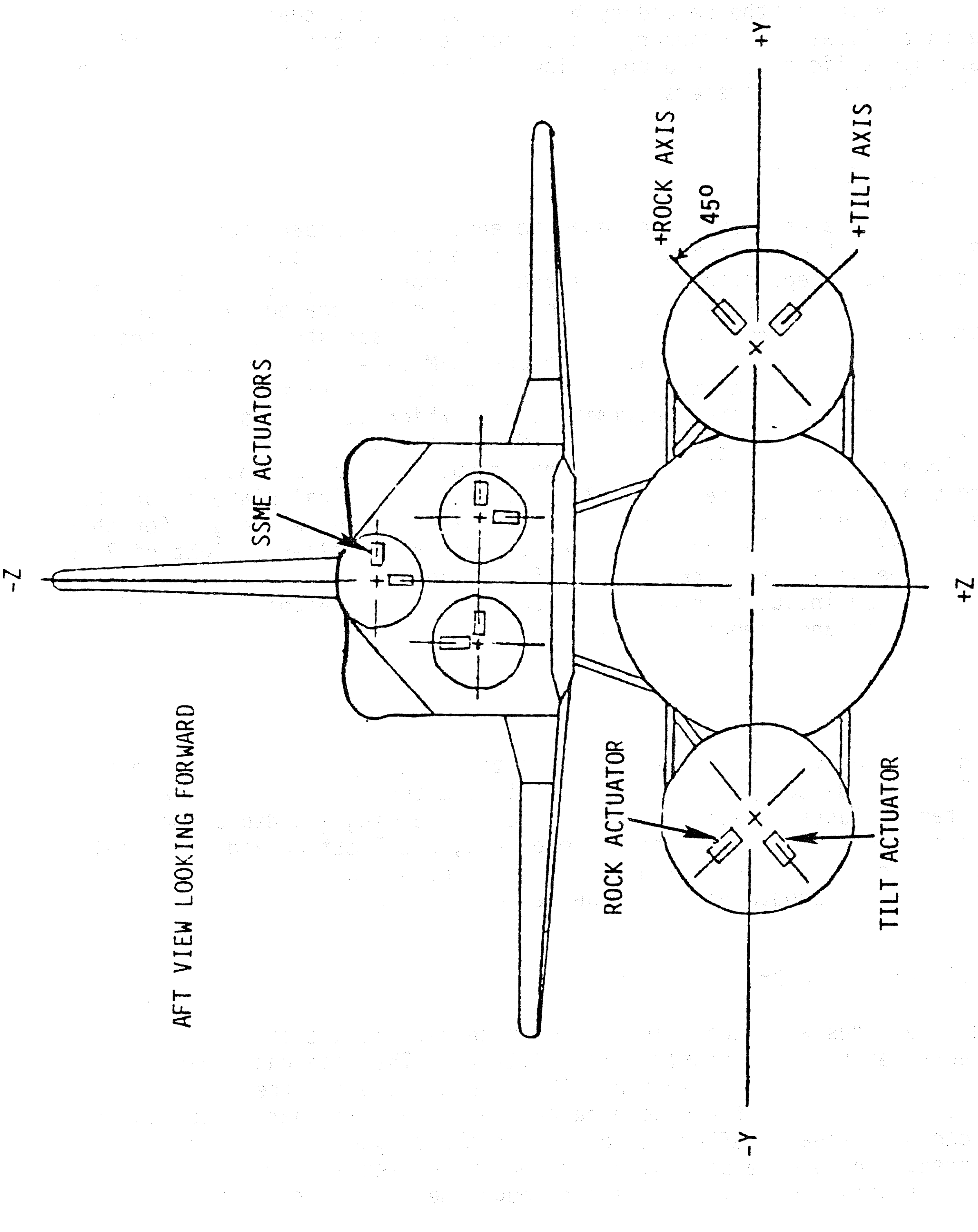


Figure 4.1-8.- SRB/SSME actuator orientation.

4.1.4.2 Hydraulic Power Units

Each SRB has two hydraulic power units (HPU's) as a part of the TVC system. On an SRB, each HPU is connected to both the rock actuator and the tilt actuator. One HPU serves as the primary hydraulics on the actuator, and the other HPU serves as the secondary hydraulics. Each actuator has a switching valve that allows the secondary hydraulics to power both actuators if the primary hydraulic pressure drops below 2050 psig (±150 psig). Details of the HPU are found in systems brief 4.6.

4.1.5 Separation

The separation subsystem is designed to ensure safe separation of each of the SRB's from the ET without damaging or recontacting the orbiter/ET, during or after separation. The separation subsystem (fig. 4.1-9) consists of a release system, sensors, and separation bolts located in the SRB/ET forward attach fitting and in each of the aft attach struts; also included are eight solid booster separation motors (BSM's) - four mounted in the SRB nose frustum and four mounted externally on the aft skirt (fig. 4.1-10). The aft motors are located unsymmetrically, which causes a small roll movement to be imparted to the SRB. The BSM's are located such that the SRB moves from the orbiter/ET in a way that reduces the plume and particle impingement on the orbiter. The BSM motor has a nominal diameter of 12.85 inches and a length of 31.1 inches. The maximum expected weight for this motor is 152 lb. At a nominal 70° F, the motor propellant weight of 70.6 lb will provide an average thrust of 22,505 lb for 0.55 second. Separation subsystem also includes an umbilical pull-away unit weighing about 675 lb. All sequencing and commands are issued by the orbiter.

4.1.6 Recovery

The booster recovery subsystem provides the necessary hardware to control the SRB final descent velocity and attitude after separation. The recovery subsystem includes parachutes, methods of sequencing and deploying these parachutes, parachute separation components, and location aids that help in search and retrieval operations for the expended booster and for the parachutes. The parachute deploy sequence is shown in figure 4.1-11.

4.1.6.1 Pilot and Drogue Parachutes

These parachutes are housed in the SRB nose cap and are deployed as the nose cap separates from the frustum (fig. 4.1-12). The nose cap separation is initiated by a barometric switch. The nose cap deploys the pilot parachute which in turn deploys the drogue parachute. The pilot parachute bag and nose cap are released after deployment of the drogue parachute and are not recovered. The drogue parachute, nominally 54 feet in diameter, stabilizes and decelerates the SRB. It opens through one reefing stage to full open.

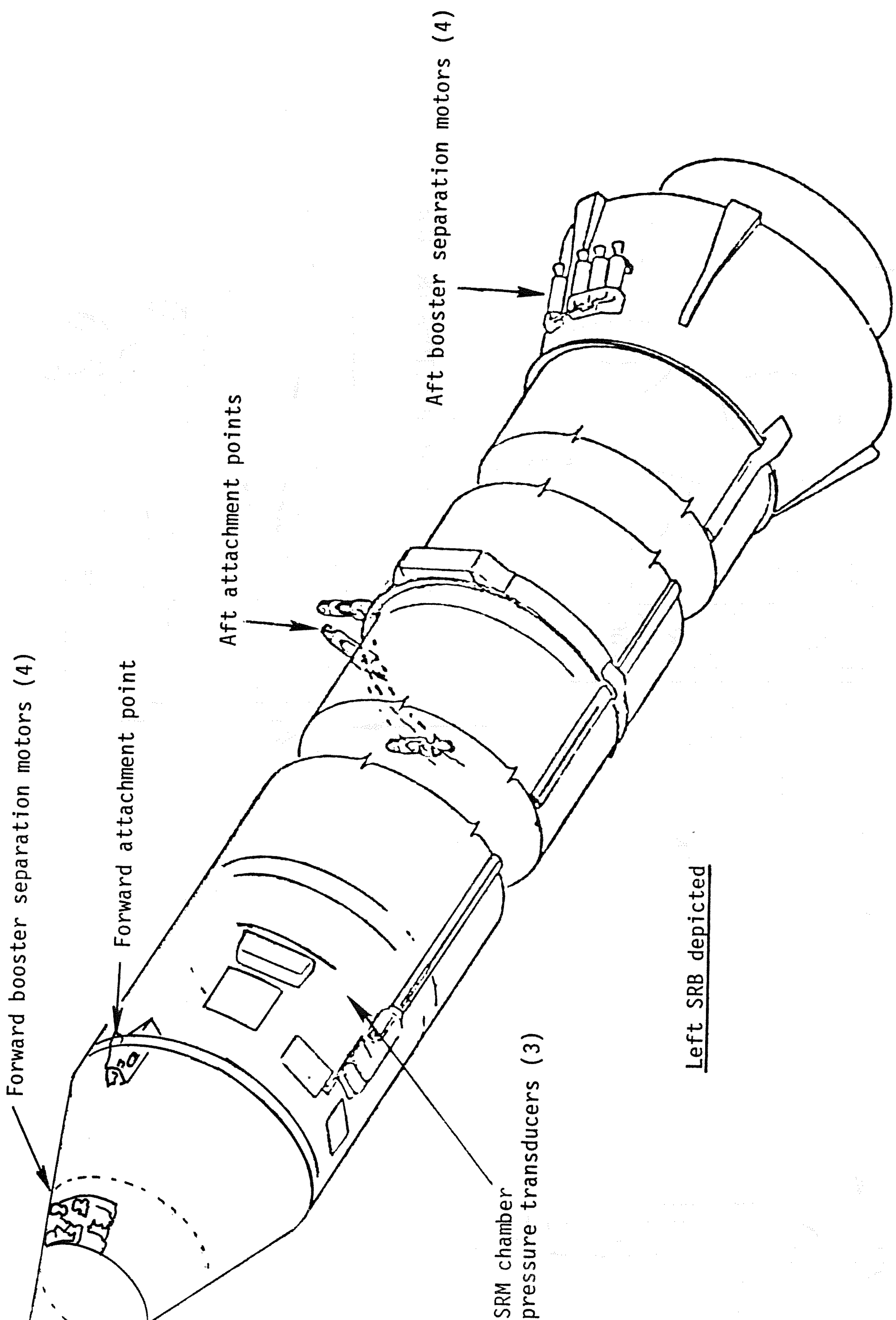


Figure 4.1-9.- SRB separation system components location.

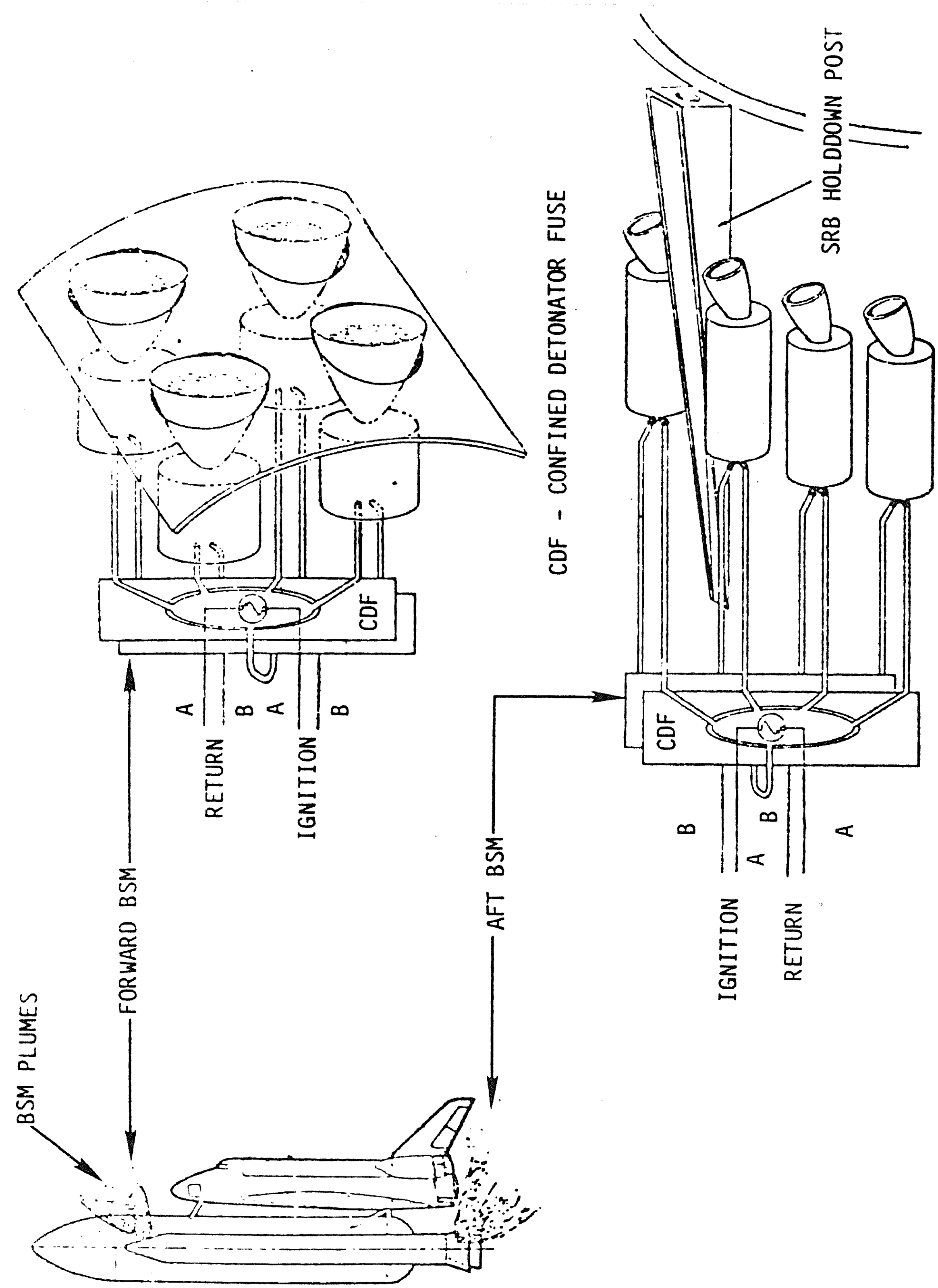


Figure 4.1-10.- SRB booster separation motors configuration.

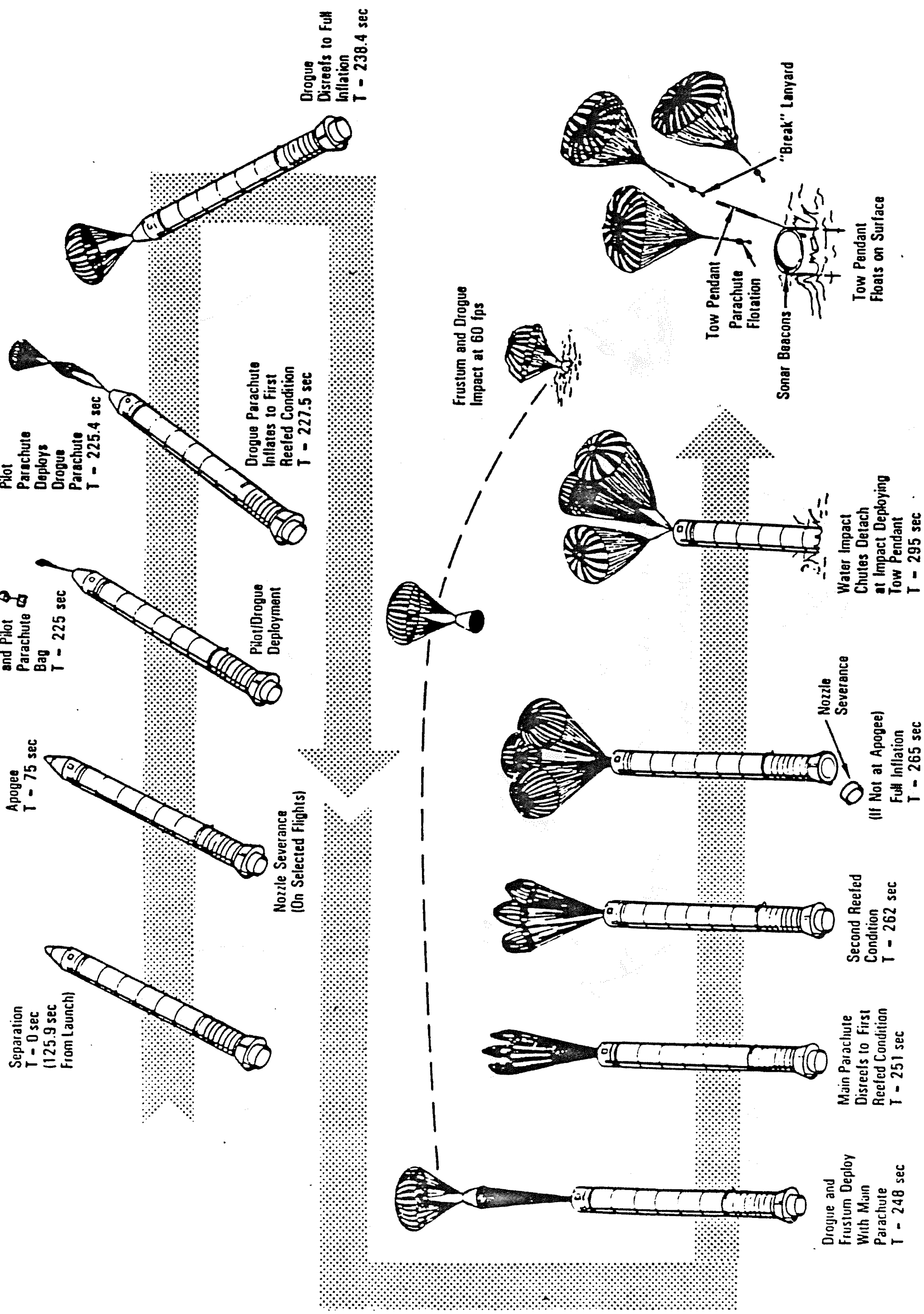


Figure 4.1-11.- SRB parachute deployment sequence.

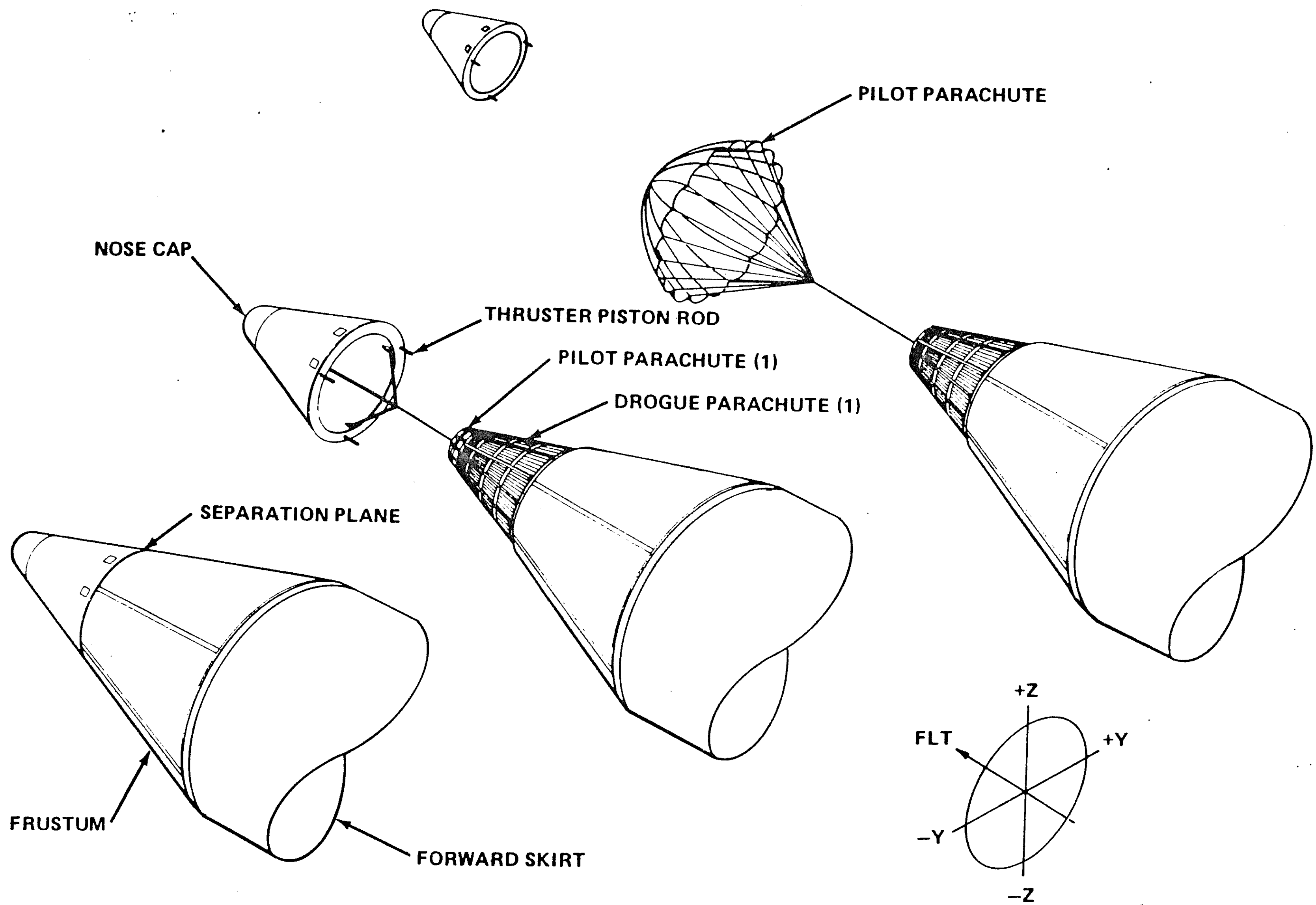


Figure 4.1-12.- SRB nose cap separation.

A second barometric switch output initiates separation of the frustum. As the drogue parachute pulls the frustum away from the SRB, the main parachutes are deployed. The drogue parachute decelerates the frustum for recovery.

4.1.6.2 Main Parachutes

The main parachute assembly decelerates the SRB to a 81-ft/sec nominal water impact velocity. The parachute cluster assembly consists of three parachutes, each approximately 136 feet in diameter, and is housed in the frustum structural component. The main parachutes are deployed as the frustum separates from the booster. The parachutes are opened though two reefed stages to full open. At booster impact, the main parachutes are disconnected and the SRB directional finding beacons and lights are actuated. The main parachutes have flotation gear and location aids to help in recovery operations.

4.1.6.3 Recovery Operations

A command is sent from the orbiter to the SRB just before separation to apply battery power to the recovery logic network. A second, simultaneous command arms the three nose cap thrusters (for deploying the pilot and drogue), the frustum ring detonator (for main parachute deployment), and the main parachutes disconnect ordnance. Two altitude switches, which sense barometric pressure, are set to close at high altitude (below 19,000 feet) and at low altitude (below 10,000 feet). At high altitude, the nose cap thrusters fire, pushing the nose cap away from the booster, which deploys a pilot parachute. This pilot parachute deploys the drogue parachute. The drogue parachute serves to orient the SRB into a nozzle-first attitude and to provide some initial deceleration of the SRB.

At the low altitude switch output, the frustum ring detonator fires, and a shaped charge frees the frustum. The drogue parachute pulls the frustum away from the booster, thus deploying the main parachutes. At booster impact the main parachutes are disconnected and the SRB radio-direction-finding beacon and light are turned on.

The nominal velocity for the booster at water impact has been set at 81 feet per second. The frustum, lowered by the drogue parachute, impacts within approximately one mile of the SRB at a nominal 50 ft/sec. The frustum provides buoyancy and location aids for the drogue parachute. The main parachutes have flotation gear and location aids. Retrieval of the boosters, parachutes, and frustums will be accomplished using surface vessels. The retrieval vessels will tow the boosters to Kennedy Space Center, other objects recovered will be brought on board the vessels. The boosters are expected to impact at a point approximately 140 n.m. downrange, ±5 n.m.. Once the boosters are located, an underwater maneuverable device is launched and remotely controlled from the retrieval vessel to plug the SRB nozzle and dewater the motor case. This causes the SRB to change from a nose-up floating position to a more horizontal attitude. The recovery assemblies are planned for 10 flights.

4.1.7 Electrical and Instrumentation

The SRB E&I subsystem for operational flights consists of two major functional systems dedicated to two specific portions of the SRB mission. One system is operational from prelaunch until SRB/ET separation and will be referred to as the ascent system. The other system is operational from just prior to SRB/ET separation until SRB splashdown and will be referred to as the recovery system. These systems are identical on both the left and right SRB's. E&I subsystem components are generally designed for recovery, refurbishment, and reuse on 19 subsequent flights or a total of at least 20 uses.

4.1.7.1 Ascent System

This system consists of the E&I subsystem components necessary to respond to Orbiter Vehicle (OV) commands for controlling SRB prelaunch functions, ignition, power ascent, and SRB separation. In addition, it provides the data acquisition element necessary to implement the OFI program. As a purely responsive system, the ascent system relies on the OV for all power and stimuli. The system is redundant in all critical functions.

4.1.7.2 Recovery System

The system consists of the E&I subsystem components that bring about the successful recovery of the SRB after burnout and separation, by performing the following functions:

- Severance of the nozzle extension
- Deployment of the drogue and main chutes
- Powering of location aids
- Severance of main chute at water impact

The recovery system is simplex and self-contained, except for initial power-on command, which is derived from the separation fire commands from the OV.

4.1.7.3 Operational Components

The E&I subsystem components can generally be divided into eight elements: the forward and aft integrated electronics assemblies (IEA's); the rate gyro assemblies (RGA's); the instrumentation sensors, SRB location aids; recovery battery; altitude switch; frustum location aids; and interconnecting cable assemblies.

4.1.7.4 Integrated Electronics Assemblies

The forward and aft IEA's are identical in many of the functions they perform. These are power distribution, command/control switching distribution, data bus termination, signal conditioning, multiplexing of SRB measurements, and demultiplexing of OV commands. The forward IEA also contains the recovery logic-sequencer and is mounted on a structural ring in the watertight forward compartment. The aft IEA is mounted in the ET/SRB attach ring. In addition to the functions listed above, the aft IEA is also used for the routing of all interface wiring from the OV and provides the control electronics for the TVC hydraulic power units.

Each IEA is divided into three elements:

- A distributor for power distribution and command/control switching and distribution. The distributor contains: the pyrotechnic initiator controllers (PIC's) for SRB ignition, ET/SRB separation and recovery; solid state switch cards; TVC APU controller card in the aft IEA; and the recovery logic sequencer and impact switch in the forward IEA.

- A signal conditioner for the conditioning of instrumentation sensor outputs that are not suitable for direct interface with the SRB MDM's.

- An MDM for multiplexing SRB measurement data to the OV and demultiplexing OV commands and interrogations directed to the SRB. The MDM is redundant and transmits and receives via two independent data buses.

4.1.7.5 Rate Gyro Assemblies

Each SRB contains two RGA's, with each RGA containing one pitch and one yaw gyro. These provide an output proportional to angular rates about the pitch and yaw axis to the orbiter computers and guidance, navigation, and control system during first stage ascent flight in conjunction with the orbiter roll rate gyros until SRB separation. At SRB separation a switchover is made from the SRB RGA's to the orbiter RGA's. The SRB RGA rates pass through the orbiter aft IEA and MDM's.

Future flights are considering placing all RGA's from the SRB's to the orbiter. This is a result of the large amount of damage that has occurred to SRB RGA's during parachute opening and water impact, prohibiting RGA reuse.

4.1.7.6 Instrumentation Sensors

Instrumentation sensors are provided to meet the requirements of the OFI program.

4.1.7.7 SRB Location Aids

The SRB location aids consist of an RF beacon and a flashing light. The beacon antenna and the flashing light are mounted on the apex of the forward pressure dome.

4.1.7.8 Recovery Battery

The recovery battery is a primary 50-ampere hour, silver/zinc battery that provides all power for the recovery system and is not designed for reuse. This battery also provides power for one of the redundant channels in the RSS.

4.1.7.9 Altitude Switch

The altitude switch assembly, mounted in the frustum area, provides initiation of drogue and main chutes, deployment, and recovery aids power on.

4.1.7.10 Frustum Location Aids

The frustum location aid is a self-contained assembly, having a flashing light, RF beacon, salt water sensing switch, and battery.

4.1.7.11 Cable Assemblies

Cable assemblies are provided for interfacing with the ET-provided cable assembly at the pullaway connectors, interconnection of ascent or recovery system components, and interconnection of the ascent or recovery system with the end item components being controlled or monitored. In general, operational flight cable assemblies are designed for reuse on subsequent flights. Some exceptions to this are: cables that have connectors that are exposed during recovery and retrieval operations and cables that are connected directly to ordnance initiators.

4.1.7.12 Range Safety System

The RSS contains the necessary equipment to destroy the SRB in case of a malfunction requiring premature flight termination. The system is ground-operated and redundant. It is monitored, tested, and safed via the MDM in the forward IEA (figs. 4.1-13 and 4.1-14). The RSS components include the following:

- RSS Distributor - Contains the PIC's and switch cards, necessary to detonate the linear-shaped charges. The distributor is powered from the DFI battery and the recovery battery. The distributor contains both redundant channels.

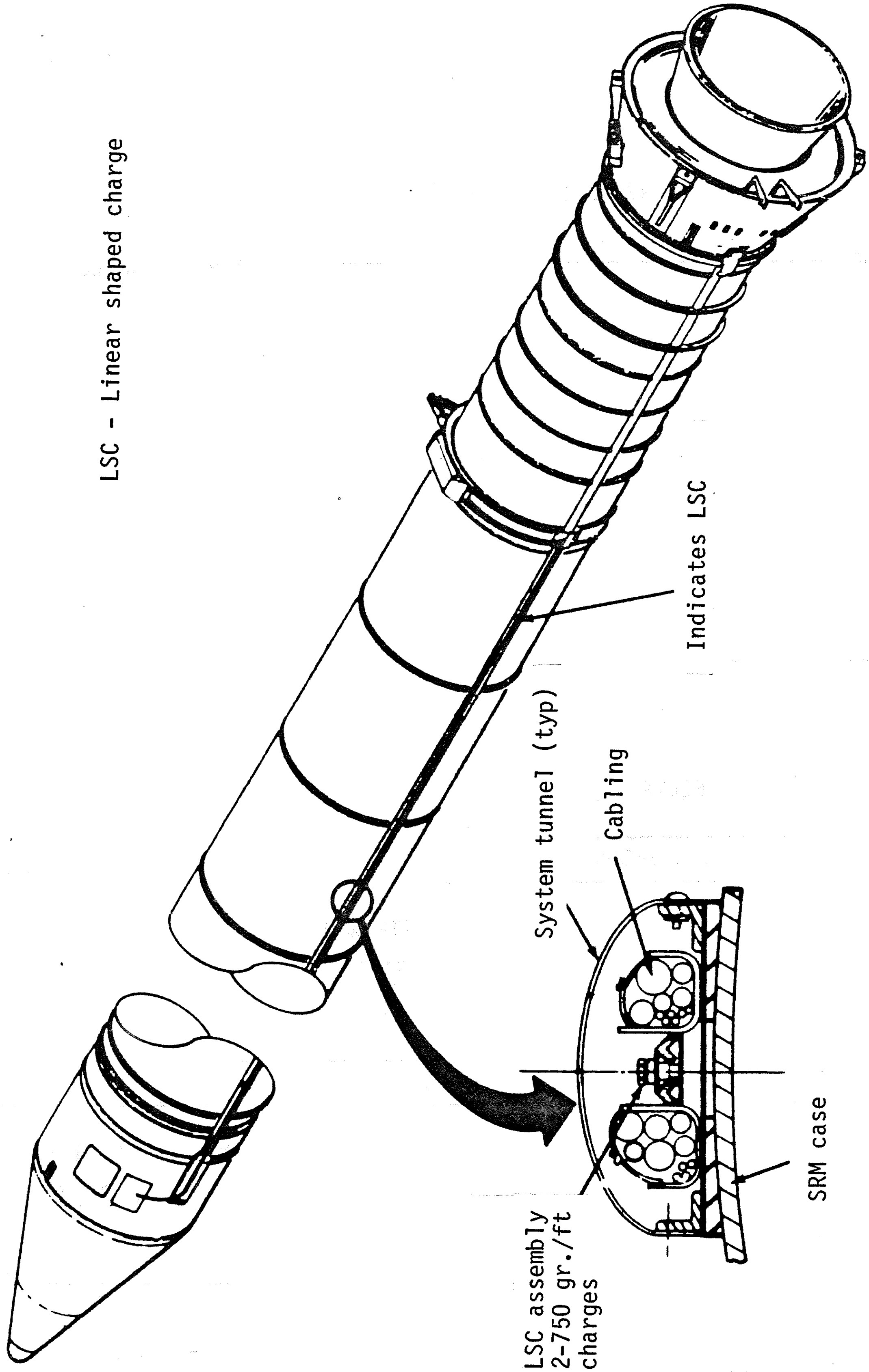


Figure 4.1-13.- SRB LSC location.

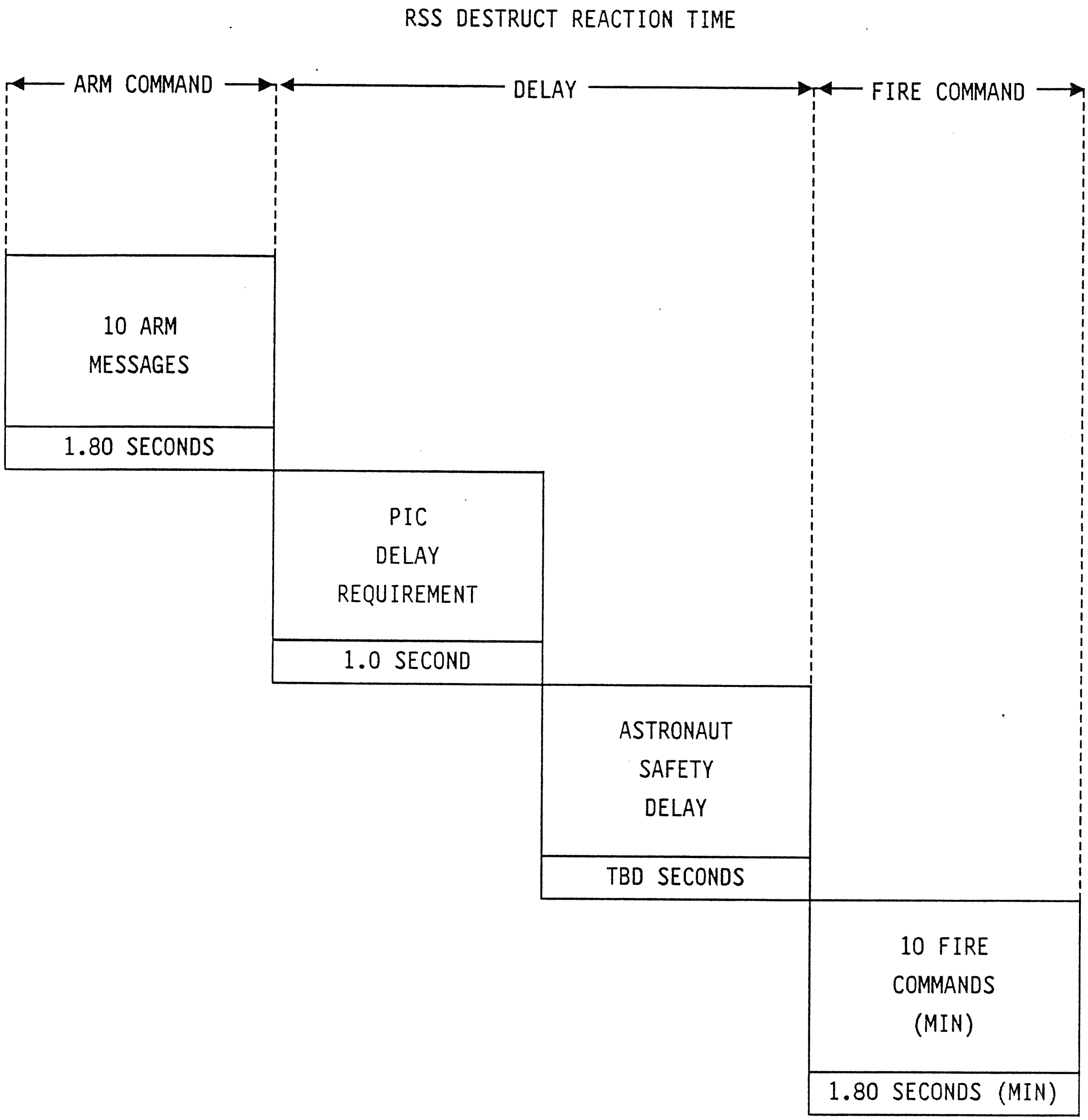


Figure 4.1-14.- RSS destruct reaction time.

- Decoder - Provides the arm and fire commands for the distributor. There are two redundant decoders.
- Receiver - Provides the demodulated message that serves as an input to the decoder. There are two redundant receivers.
- Couplers - There are two couplers for the antenna - one is a directional and the other a hybrid coupler. The hybrid coupler provides a ground input to the receiver for test purposes. The couplers have redundant channels.
- Antennas - There are two antennas mounted 180° apart on the forward skirt.
- S&A - The safe and arm (fig. 4.1-15) device.

4.1.8 Mission Operations (fig. 4.1-16)

The times given below could vary flight-to-flight. These times are for information only.

4.1.8.1 Prelaunch

Time*	Events**
To - 10:00:00	SRB aft skirt GN2 purge activation
T - 9:00:00	SRB buses A and B power applied (PC1 & PC2 active)
To - 5:00:00	Check LH and RH SRB TVC SYS A and B FSM pressure (350 psia minimum prior to T-31 sec)
T - 8:00:00	SRB field joint heaters enabled
To - 3:00:00	Final SRB bus C power applied (PC3 active)
T - 1:30:00	Check SRM Pc transducer calibration
T - 1:00:00	Turn on SRB RGA's for pretest warmups
T - 0:35:00	SRB RGA tests
To - 0:30:00	RSS checks (RSS inhibit - on)
To - 0:29:00	Turn ATVC power on (ATVC sw (4) - ON)
To - 0:09:00	Start auto launch sequence (LPS) Verify igniter heater off and field joint heaters are off

Time*	Events**
T_O - 0:05:00	Arm RSS S&A device (3) Arm SRM ignition S&A devices (2)
T_O - 0:00:30	Issue SRB actuator OVERRIDE commands
T_O - 0:00:25	Transfer launch control from LPS to RS LAUNCH SEQ
T_O - 0:00:20	Start SRB HPU's
T_O - 0:00:16	Remove SRB actuator OVERRIDE commands
T_O - 0:00:12	Start monitoring PIC voltages HOLD if PIC voltage drops below F 37.5 V dc
T_O - 0:00:07	Remove RSS inhibit
T_O - liftoff	• Arm PIC's • SRM ignition • Holddown release • T-0 umbilical release • ET vent arm release

*Time in hr; min; sec
**Based on STS-1 with 70° F propellant

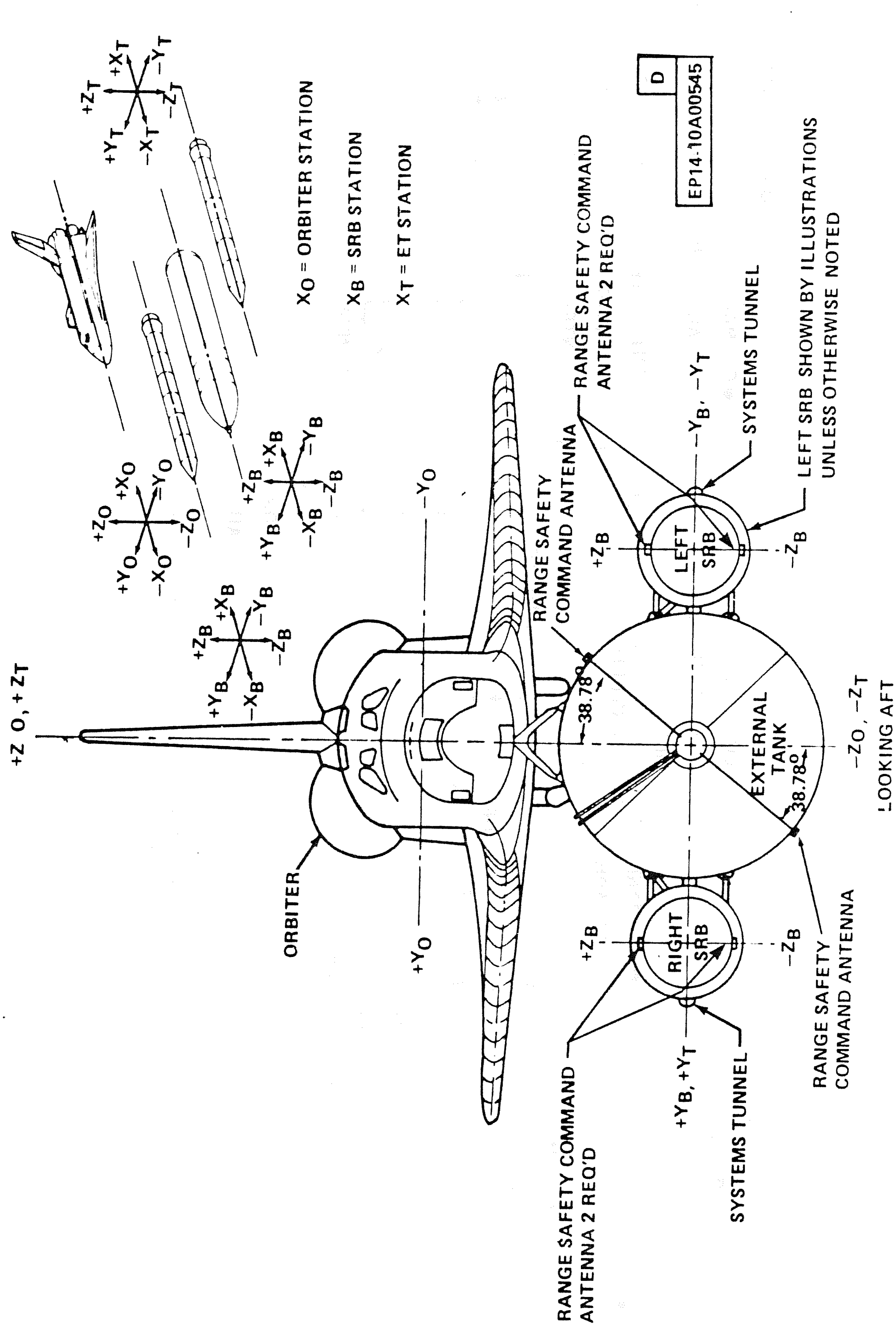


Figure 4.1-15.- RSS antenna locations.

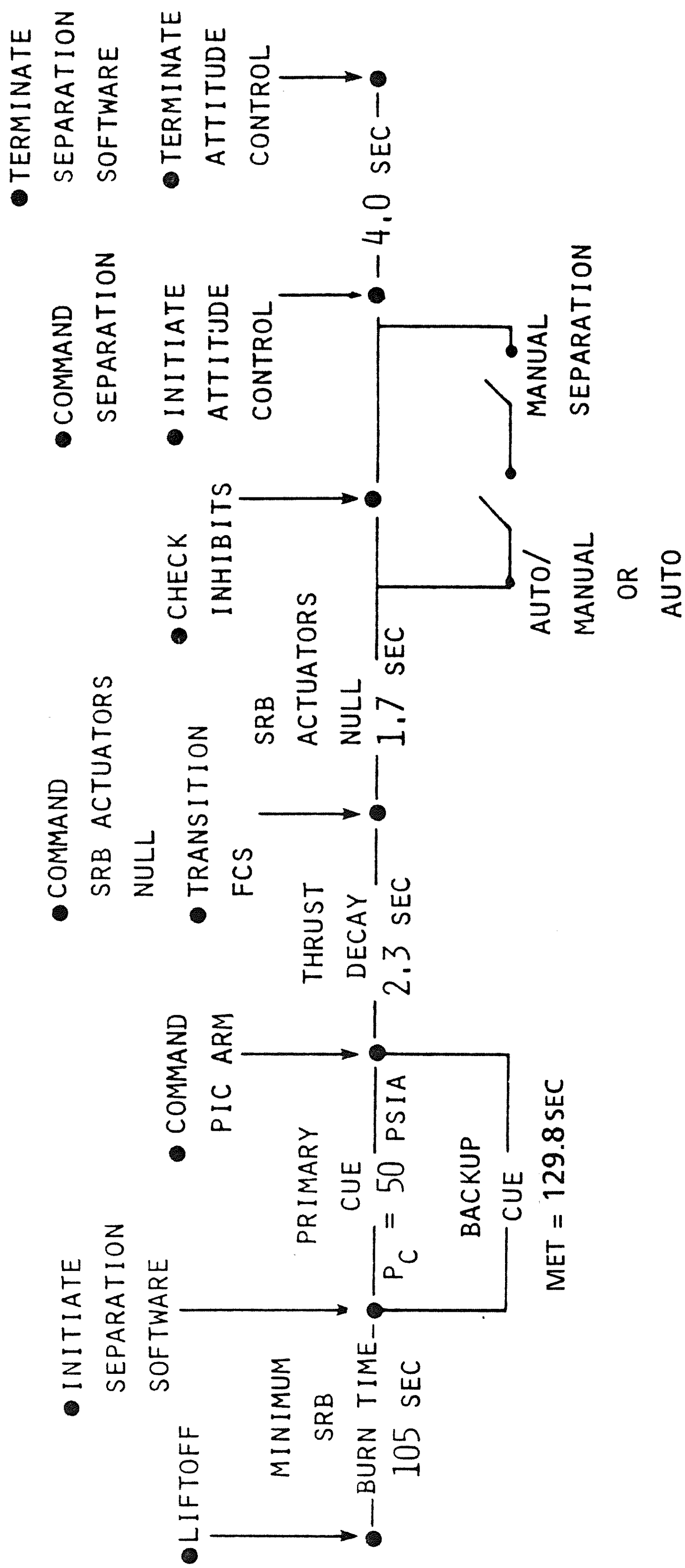


Figure 4.1-16.- SRB separation overview.

4.1.8.2 Ignition

Time*	Events**
T_O - 0:00:06.5	Issue SSME START ENABLE command
T_O - 0:00:06.6	Issue SSME START command PAD SHUTDOWN if: (1) PIC voltage < 37.5 V dc (2) Launch sequence abort flag set (3) Any SSME not > 90 percent Pc within 4 sec
T_O - 0:00:06	SRB systems ready for launch
T_O - 0:00:00	SSME (3) > 90 percent Pc (nominal)
	Issue SRB actuator override commands
	Issue SRM ignition FIRE 1 command
	Issue SRM ignition FIRE 2 command (defines T_O) • Ignites SRM's • Blows holddown bolt NSI's • Resets MET to zero (T_O)
T_O + 0:00:00.04	Issue T-0 umbilical release FIRE 1 command
	Issue T-0 umbilical release FIRE 2 command
	Remove SRB actuator OVERRIDE commands

4.1.8.3 First Stage Ascent

Time*	Events*
T_O + 0:00:19.7	Maximum SRM thrust
T_O + 0:00:35.5	SSME throttle-down to thrust bucket
T_O + 0:00:63	SSME throttle-up to nominal power level
T_O + 0:00:69	Maximum dynamic pressure (q)
T_O + 0:01:40	Start monitoring Pc and MET

*Time in hr; min; sec
**Based on STS-1 with 70° F propellant

4.1.8.4 SRB Separation

Time*	Events**
T_O + 0:02:00	Pc < 50 psia on both SRB's or when MET = 129.8 sec If Δt between LH and RH SRB's is > 5.5 sec, then go to MET < 129.8 sec cue Arm SRB separation PIC's Configure systems for separation
T_O + 0:02:04	Check for manual override of sep inhibits (crew controls) Check SRB separation inhibits Out-of-limits separation inhibits: (1) Stops separation sequence at this point (2) "SEP INH" appears on TRAJ (PASS) and TRAJ 1 (BFS) CRT displays (3) Requires manual override via crew controls (SEP sw and pb on panel C3) if separation inhibit parameter does not come back within limits
T + 0:02:06	Issue SRB SEP FIRE 1 and FIRE 2 commands

```
*************************
*                       *
* SRB SEPARATION OCCURS *
*                       *
*************************
```

*Time in hr; min; sec
**Based on STS-1 with 70° F propellant

4.1.9 Advanced Solid Rocket Motor

In the future, the current space shuttle RSRM's will be replaced by a new, redesigned advance solid rocket motor (ASRM). The ASRM, with its enhancements, will provide improved reliability and performance. The ASRM will be produced by Lockheed Missiles and Space Company in conjunction with Aerojet and Thiokol.

The ASRM will be 1513.5 in. in length and 150 in. in diameter, increased from the current 146 in. diameter. The average thrust vacuum is rated at 2,624,031 lbf., and delivered Isp in a vacuum is 270.3 seconds. The average chamber pressure is 633 psia. Action time is predicted to be 134.13 seconds.

The ASRM case features a three-segment design with two field joints and welded factory joints. The field joints are axially bolted, a feature that is expected to result in improved safety and reliability over the current pin and clevis configuration. Axially bolted joints should be easier to inspect, defective fasteners should be easier to replace in the configuration, and they should allow for easier assembly and disassembly.

The case also features integral stiffeners and an integral external tank attach ring. These are expected to eliminate failure points that were associated with the bolt-on stiffeners and ET attach ring.

Improvements to the nozzle design include the elimination of two out of the five internal joints, the reduction of flame and nonflame insulator joints, and the reduction of its erosion rate by the use of a carbon-carbon composites.

Improvements were also made to the insulation and ignition to produce what are believed to be significant safety and reliability enhancements.

4.2 SRB CASING

4.2.1 General

The SRM case functions as a pressure vessel in which thrust can be developed and as a structural frame through which flight loads are transmitted and reacted. The case is designed to withstand ascent, reentry, and water impact loads and to be reusable after refurbishment and propellant loading. The case is designed for a maximum expected operating pressure (MEOP) of 1016 psia at 90° F. The case construction is a D6AC steel forging with no welds forming each of 11 segments. These segments are designated as the forward dome segment, six cylindrical segments, the ET attachment segment, two stiffener segments, and the aft dome segment.

4.2.1.1 Forward Segment

The forward segment is swaged from a pancake billet of D6AC steel material. The use of a relatively flat 1.6:1 ellipse allows for a completely weld-free part. The dome thickness is tapered to maintain a constant stress. A short stub skirt with a male (tang) interface has been provided for attaching the SRB forward skirt assembly. A 21-inch inside diameter (ID) polar boss with bolted attachment provisions is provided at the apex of the forward dome for attachment of the ignition system. Forty 0.75-inch bolt holes are provided for this purpose. An aft facing, male (tang) joint, has been incorporated for attachment of this segment to the cylinder case segment.

4.2.1.2 Cylindrical Segment

The six cylindrical segments are roll-formed from rolled ring forgings. The basic cylindrical wall is roll-formed "net" and no additional machining is required. Tang and clevis joints at either end are machine-finished after heat treatment in order to achieve the close tolerances required. In the past, all cylinder segments were interchangeable since they all shared the same joint design. The configuration of factory and field joints was the same. With the incorporation of the redesigned field joint, the interchangeability has been limited to segments sharing the same joint configurations. The basic wall thickness is nominally 0.479 inch with a tolerance of ±0.020 inch. The final nominal wall thickness after 20 uses is expected to be 0.470 inch (0.450-inch minimum).

4.2.1.3 SRB/ET Attachment Segment

The attachment segment is roll-formed from a rolled ring forging. This segment contains two attach flanges symmetrically located about motor station 1511 for attaching the SRB/ET aft attach ring. This segment has tang and clevis joints on either end for attachment to the adjacent cylindrical segment and stiffener segments. The attachment stubs and clevis joints are machined after heat treatment.

4.2.1.4 Case Stiffener Segment

The two case stiffener segments are roll-formed from rolled ring forgings. This segment contains attach flanges for attaching two circumferential stiffener ring assemblies. The function of the stiffener rings is to prevent case buckling due to the cavity collapse loads during water impact. Only the aft stiffener segment has externally attached rings. The flanges provide sufficient buckling stiffness for the forward segment. The segments are identical and interchangeable.

4.2.1.5 Aft Segment

The preform for the aft segment is swaged into a hemispherical shape from a cylindrical rolled ring forging. A short stub skirt, with a male (tang) interface, is provided for attaching the SRM to the SRB aft skirt assembly. A polar boss with bolted attachment provisions for the nozzle assembly is provided on the aft end. One hundred 1.375-inch bolt holes are provided at at bolt circle diameter of 106.1 inches. Nominal wall thickness of the hemispherical aft dome is 0.362 inch with a ±0.020-inch tolerance.

4.2.1.6 Stiffener Ring

Two stiffener rings are attached to the aft case stiffener segment to prevent case buckling instability during water impact. Each ring assembly consists of three 120° sections. These sections are formed into complete 360° segments after attachment to the case by the use of splice plates. The material used for the stiffener rings is the same as that of the case in order to preclude any corrosive action due to dissimilar metal. Like the case, the stiffener rings are coated with Rust-Oleum protective coating and are reusable through a minimum of 20 times.

4.2.1.7 Factory Joints

The factory joints of the SRB casing (fig. 4.2-1) are tang- and clevis-type joints. The factory joints connect the case segments in the forward segment, the two center segments, and the aft segment. There are seven factory joints on the SRB casing. The factory joints employ two flurocarbon "viton" O-rings, a pin retainer band, a leak check port, and a vulcanized weather seal. Additional insulation plies are layered on the interior to insure joint integrity.

4.2.1.8 Field Joints

The SRB case incorporates three field joints into its design (fig. 4.2-2). These joints are where the four case segments are joined during vertical assembly at Kennedy Space Center. The field joints are also tang and clevis-type, but have several additional features, compared to the factory

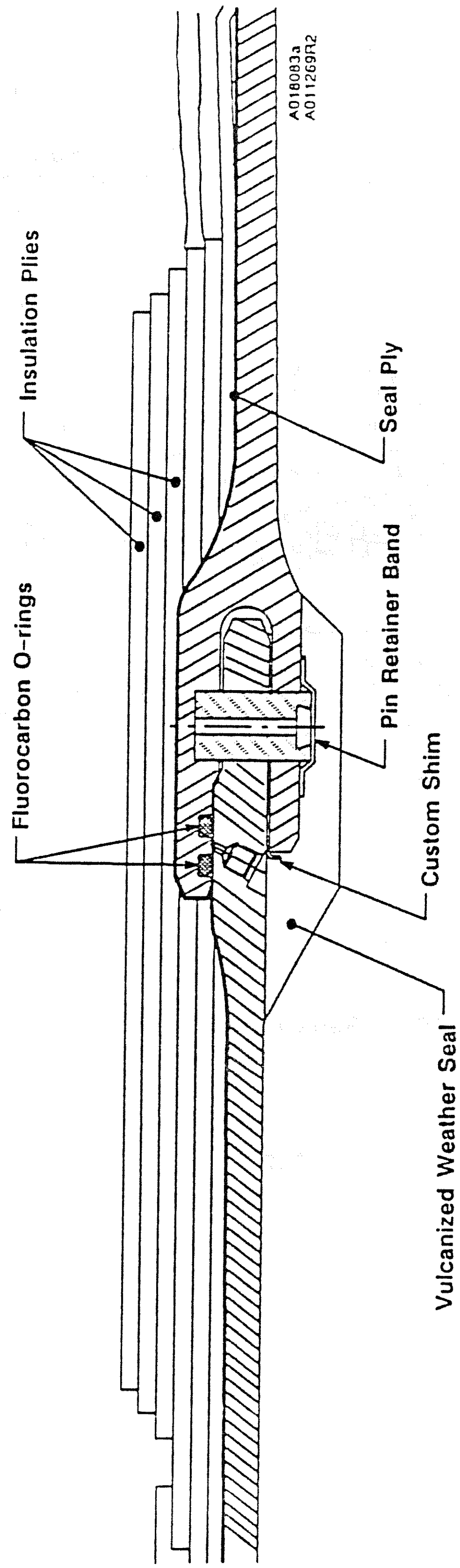


Figure 4.2-1.- Factory joint.

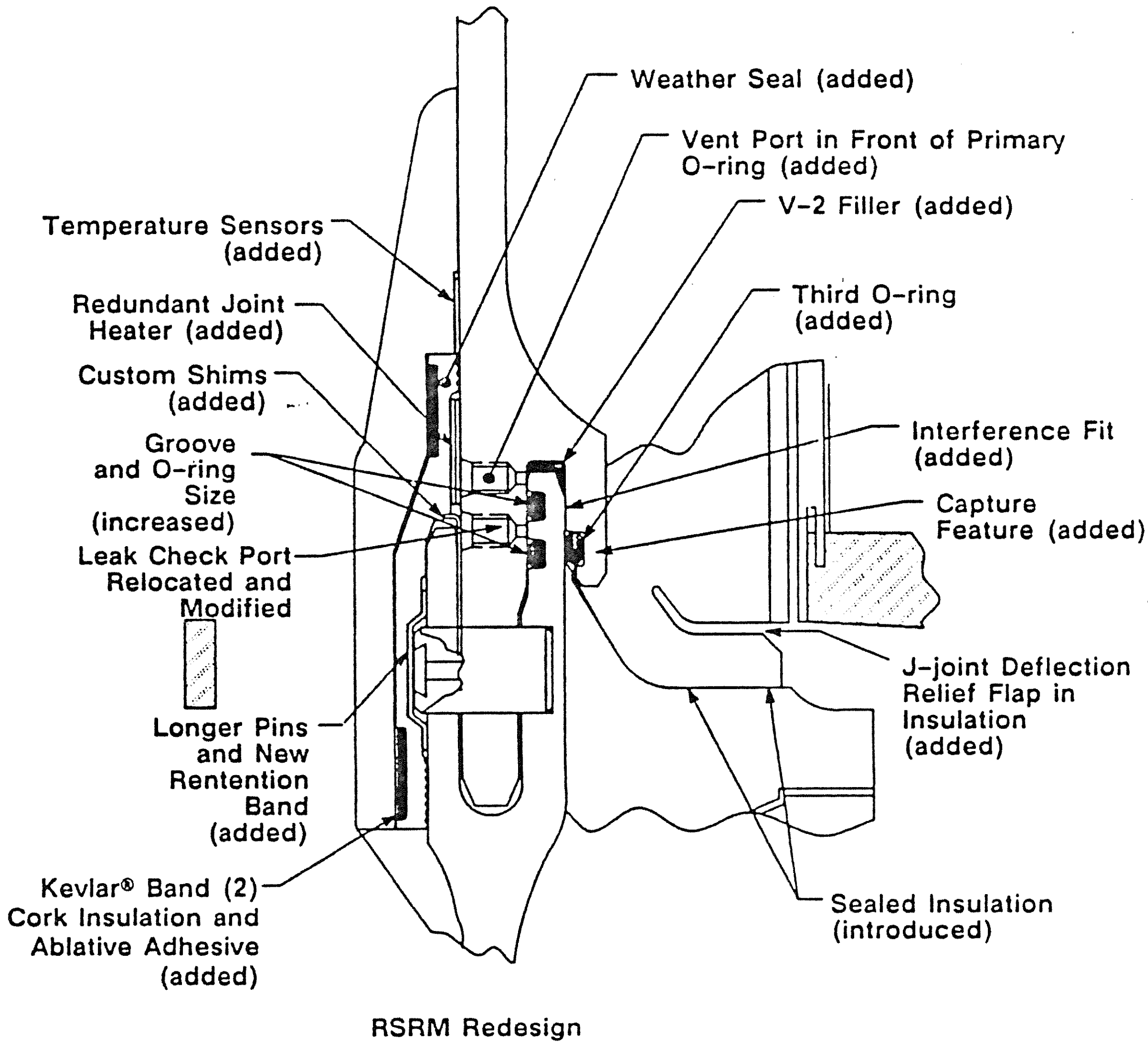


Figure 4.2-2.- Field joint.

joints, to protect against any type of hot gas leak. The most notable difference is the tang capture feature. This is intended to provide a positive metal-to-metal interference fit around the circumference of the tang and clevis ends of the joined segments. The interference fit limits the deflection between the tang and clevis O-ring sealing surfaces caused by motor pressure and structural loads. The joints are designed so that the seals will not leak under twice the expected structural deflection. The field joints also have three O-rings, with the additional one in the capture feature. There are also two leak check ports, and a pin retention band. The joints also have external heaters with integral weather seals which were incorporated to maintain the joint and O-ring temperature at a minimum of 75° F. The seals also prevent water from penetrating into the joint. The internal case insulation for the joints features a J-seal; a pressure actuated flap.

4.2.1.9 Case to Nozzle Joint

These joints (fig. 4.2-3) incorporate 100 radial bolts, and 100 axial bolts to minimize joint gap opening during pressurization. The unvented joint involves filling the joint with an ambient cure adhesive and allows the stress relief flap to account for growth variations with the joint due to temperature and motor pressurization. The joint also contains a wiper O-ring designed to keep adhesive away from the primary seal during assembly. Vent ports and slots vent assembly pressure and position the primary O-ring.

4.2.2 Functional Description

4.2.2.1 SRM Case

The 11 case segments are preassembled in four subassemblies prior to propellant casting. These four subassemblies are designated as the forward casting segment, the center casting segment (two required), and the aft casting segment.

The forward casting segment consists of a forward segment and two cylindrical segments. The two center casting segments each consist of two cylindrical segments, and the aft casting segment consists of an attach segment, two stiffener segments, and the aft segment.

The individual segments are joined with pinned clevis joints and pin retrainer bands which hold the pins in place and also provide environmental seals for each joint. The clevis joints utilize 180 cobalt alloy pins. Three of these pins, located at approximately 120 degree intervals, are used for case alignment during assembly. The remaining 177 pins are load bearing and after assembly are held in place by the retainer bands. Attachment of the ignition system and the nozzle is by means of polar bolt circles in the forward and aft segments, respectively. The igniter components in the forward dome use Gask-O-Seals and are dual sealing surfaces. Pressure seals at each segment joint are dual O-rings.

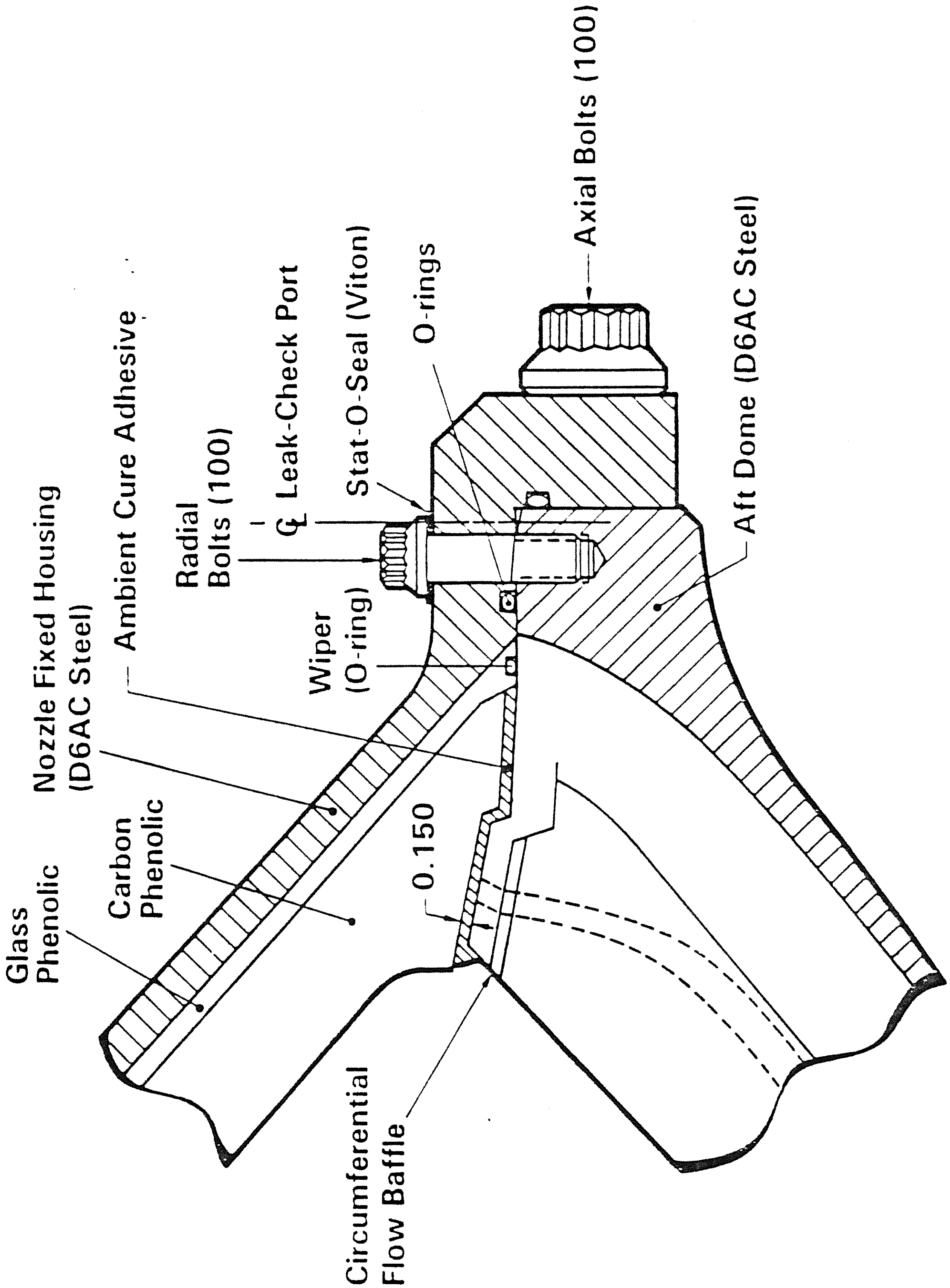


Figure 4.2-3.- Nozzle-to-case joint.

The assembled SRM case has a nominal overall length (boss to boss) of 1388.77 inches. The outside diameter (OD) of the basic cylindrical wall is 146.000 inches and the maximum OD of the metal parts is nominally 149.478 inches at the attach stub. The total empty case weight is 98,010 lb. The external surface of the assembled case is protected from general corrosion and stress corrosion by the Rust-Oleum paint.

SRM standard case configuration is shown in figure 4.2-4.

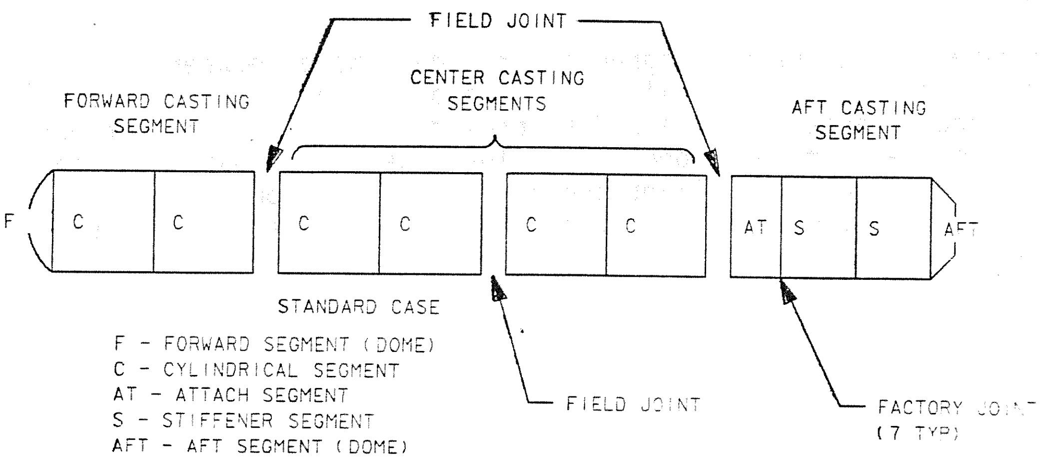


Figure 4.2-1.- SRM standard case configuration.

4.2.2.2 Insulation

The insulation subsystem includes chamber insulation, propellant relief flaps, and forward and aft facing inhibitors. The insulation material used for the chamber, relief flaps, and forward inhibitors is an asbestos-silica filled nitrile butadiene rubber (NBR). The aft inhibitor material is an asbetos-filled carboxyl terminated polybutadiene polymer (HC polymer).

The insulation configuration has been designed to protect each casting segment during motor operation. The internal insulation subsystem for each casting segment includes the primary insulation, forward facing full web propellant grain inhibitors, and propellant grain stress relief flaps. The primary chamber insulation propellant stress relief flaps at the aft end of each of the forward and center segments are provided to reduce insulation-liner-propellant bondline loads induced at propellant grain termination surfaces following propellant cure, thermal shrinkage, and during SRM pressurization.

The insulation components (i.e., primary insulation, propellant stress relief flaps, and forward full web inhibitor) are fabricated as integral assemblies within each casting segment. These components are laid up in the SRM casting segment, bonded, and autoclave-cured in one operation.

The inhibitors must provide thermal protection to the propellant grain and prevent the propellant from igniting and burning perpendicularly to the inhibitor surface. The forward facing full web propellant inhibitor is fabricated as an integral part of the casting segment insulator. The aft facing partial web inhibitor will be cast and trowelled on the aft face of the propellant during propellant cure. The inhibitor materials must provide a chemically compatible stratum to which the liner/propellant is bonded.

4.2.2.3 Liner

The liner material has been designed to provide a bond between the insulation and propellant in solid rocket motors. The liner is designed, formulated, and selected for the SRM to provide a liner/propellant bond that will fail cohesively in the propellant. The liner is compatible with the selected insulation and propellant and has acceptable bond throughout the service life of the SRM. The selected liner material is an asbetos-filled CTPB polymer.

4.3 SRM PROPELLANT

4.3.1 Propellant Formulation

The SRM propellant is a composite-type solid propellant. It's basic formulation is polybutadiene acrylic acid acrylonitrile terpolymer (HB polymer) with an epoxy curing agent, ammonium perchlorate, and aluminum powder. A small amount of burning rate catalyst (iron oxide) is added to achieve the desired propellant burning rate. The basic propellant consists of 86 percent total solids with the following chemical breakdown:

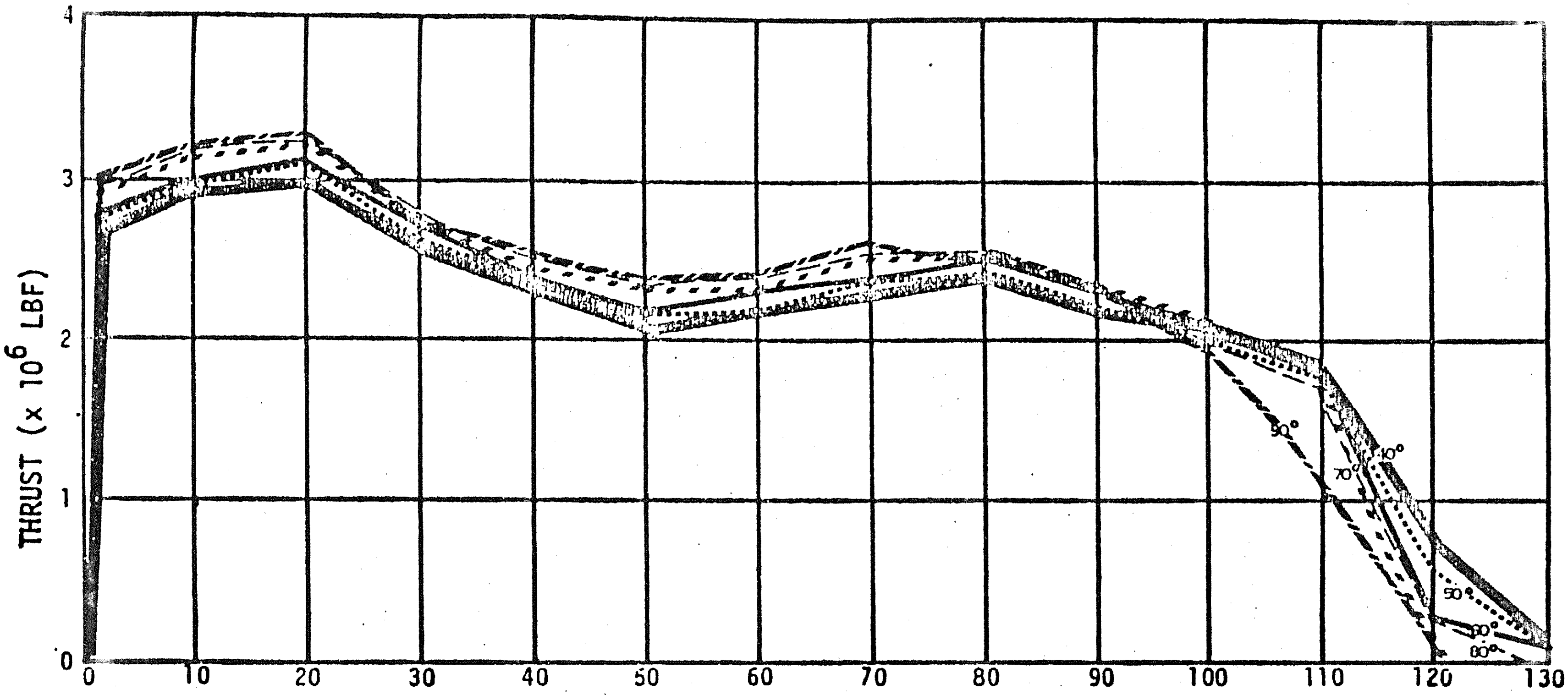
69.8 percent ammonium perchlorate (AP)
16.0 percent aluminum powder
00.2 percent iron oxide (Fe_2O_3)

The remaining 14 percent of the propellant is a combination of the HB polymer and the epoxy curing agent.

This type of propellant is usually referred to as PBAN propellant after the HB polymer, which is used as the binder.

4.3.2 Grain Design

The amount and design of the solid propellant in each segment of the SRM has been tailored to fulfill the performance requirements of the Shuttle program. These requirements are defined by the thrust-time curve as shown in figure 4.3-1. The propellant grain design consists of a forward segment with an 11-point star, which transitions into a cylindrical perforated (CP) configuration in the cylindrical portion of the segment, two identically-configured center segments which are tapered CP configurations, and an aft segment with a dual taper CP configuration. The aft face of the forward segment, both ends of the center segments, and the forward face of the aft segment are inhibited to a prescribed pattern to achieve the required thrust-time profile. The high thrust level required during the lift-off portion of the shuttle flight results from the 11-point star configuration in the forward segment. After the lift-off portion of the flight, the thrust is reduced with the burnout of the star sliver which in turn constrains flight dynamic pressure. The thrust is then progressed from the continued burn of the CP configurations in all the segments. Nearly full-inhibiting of all exposed end faces was required to achieve progressive burning off all segments (forward faces are fully inhibited while aft faces are partially inhibited). The thrust decay is designed to limit vehicle acceleration. It is achieved as the triple taper bore of the aft segment burns out. The linear 10-second thrust decay (tailoff) is produced as the final burnout of slivers in all four segments is occurring. Stress relief flaps are provided in the dome of the forward segment and in the aft end of each segment to prevent possible propellant pullaway due to differential thermal contraction during postcure and cooldown.



Event	Time (sec)	Pc(psia)	Thrust(lb)
Lift-off	0.3	546	1,900,000
Max thrust	19.7	864	3,208,330
SSME 65%	35.5	684	2,595,608
Min thrust	49.3	597	2,297,193
SSME 100%	63	628	2,423,440
Max Q	69	642	2,506,165
Begin tailoff	111	449	1,793,720
Pc = 50	120	49.5	198,931
End tailoff	120.9	25	100,000
Burnout	123.1	2.8	11,102
Separation	124	~0	~0

Figure 4.3-1.- SRM nominal thrust profile (70° F).

4.3.3 Manufacturing Process

Prior to mixing and casting of the propellant into the casting segments, definite operations must be performed. These operations are the standardization and verification of the propellant.

4.3.3.1 Standardization

Standardization is the process for characterizing designated lots of raw materials in order to select the amounts of propellant ingredients that will give the desired (target) ballistic and mechanical properties going into an SRM flight set.

Sufficient raw materials are purchased to load two SRM's (one flight set) from one standardized lot of ingredients (≃2.3M lb total). Prior to any use, the propellant ingredients are subjected to acceptance testing. Tests for specific ingredients are shown in table 4.3-I.

TABLE 4.3-I.- SRM PROPELLANT FORMULATION

Ingredient	Weight percent	Typical raw material acceptance tests
HB polymer (terpolymer of butadiene, acrylic acid and acrylonitrile)	14	Viscosity, acid content, moisture, antioxidant content
Epoxy curing agent (liquid epoxy; bis-phenol A - epichlorohydrin type)		Epoxy content, viscosity moisture
Aluminum Atomized aluminum powder	16	Chemical properties, particle size dist
Ammonium perchlorate	70 (NH_4CLO_4)*	Particle size dist, chemical properties, moisture
Iron oxide, Fe_2O_3	(≃ 0.20)*	Specific surface, chemical properties

*Determined by standardization

Once the acceptance testing of the propellant materials is complete, the standardization of the propellant can be started. The purpose of the standardization is to determine the amount of iron oxide (Fe_2O_3) required to give the required target burn rate (in/sec) and the amount of HB polymer to give the required mechanical properties. A minimum of five 5-gallon propellant mixes are made at the levels of Fe_2O_3 and HB polymer that are selected, based on historical data, to bracket the target properties, as shown in figure 4.3-2.

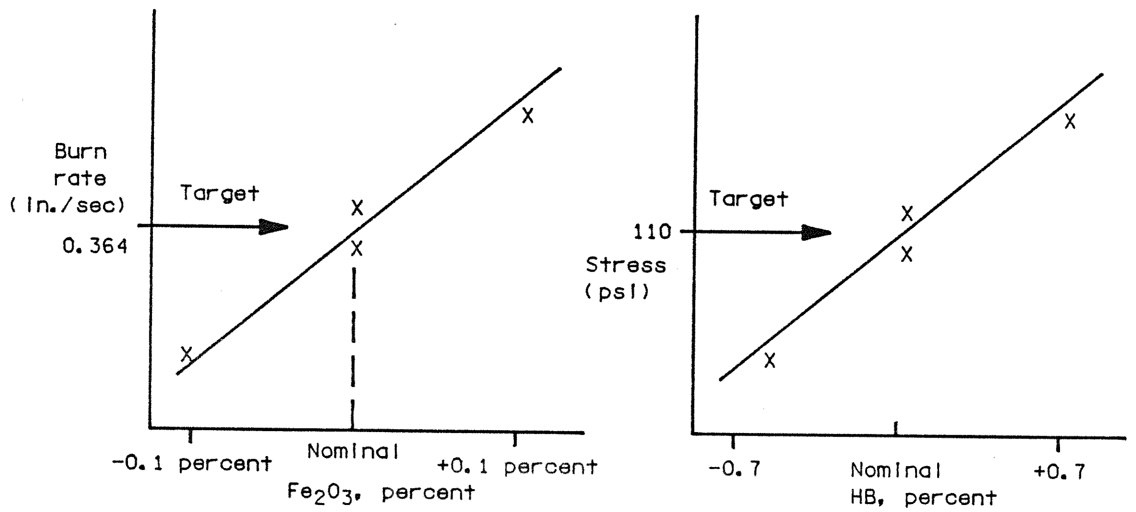


Figure 4.3-2.- Standardization target properties.

To determine the burn rates, three 5-inch CP's are cast from each of the 5-gallon mixes. (The 5-inch CP is a 5-inch od ballistic test motor having a 3-inch-diameter CP in a ≃9-inch long grain containing ≃7 pounds of propellant. The 1-inch web burns for ≃3 seconds.) These 5-inch CP's are fired, and then the burn rate is measured and plotted as in figure 4.3-2. The target burn rate in the 5-inch CP's is related to the SRM target burn rate by a scale factor (SF):

$$SF = \frac{SRM\ target\,Rb}{5\ inch\,CP\ target\,Rb}$$

where Rb is the burn rate

The SF used in a particular standardization to determine the 5-inch CP target burn rate is the arithmetic average of SF's for development motor 3 (DM-3) through the current fired flight set of SRM's.

To determine the mechanical properties, three 0.5 gallon ice cream cartons are cast and cured. The lab then tests the cured propellant for mechanical properties, which are plotted as shown in figure 4.3-2.

From the above curves the percentage of Fe_2O_3 and HB polymer that gives the desired burn rate and mechanical properties for these lots of materials may be selected.

In addition to the 5-inch CP's and the ice cream cartons that are cast from each 5-gallon mix, a 1-pint sample is taken for in-process tests.

4.3.3.2 Verification

Once the amount of Fe_2O_3 and HB polymer is determined by standardization, verification is required. This is accomplished by making a minimum of four full-scale (600 gallons, 7000 pounds) mixes of propellant at the selected Fe_2O_3 and HB polymer amounts. From each 600-gallon mix, the following are cast, cured, and tested:

- Six 5-inch CP's - for burn rate
- Three 0.5-gallon ice cream cartons - for mechanical properties
- One 1 pint - for in-process tests

From these verification mixes, the Fe_2O_3 and HB polymer content for the full scale mixes are selected to give the target properties in the SRM (fig. 4.3-3).

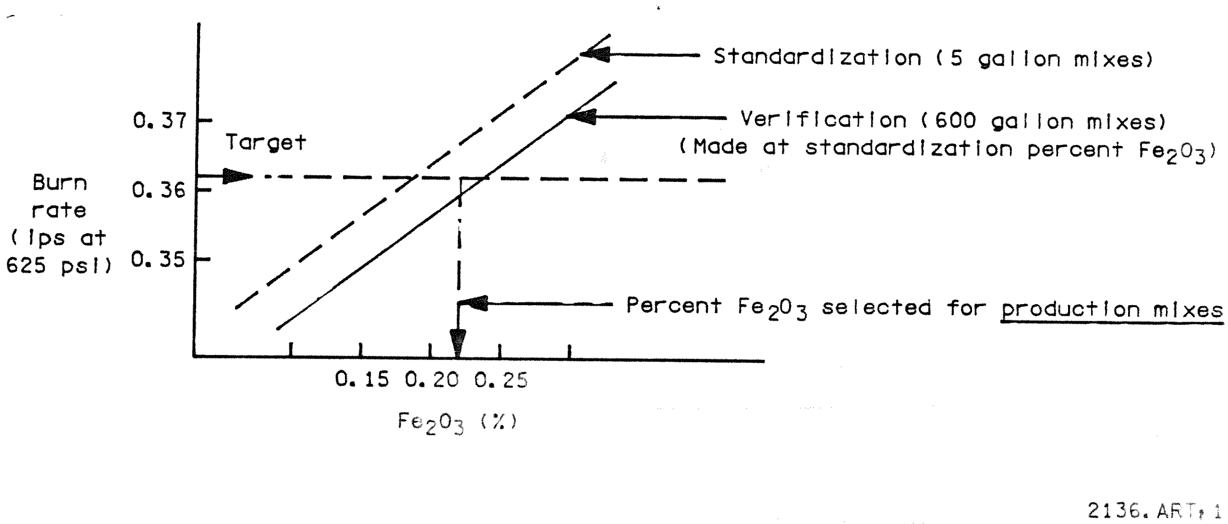


Figure 4.3-3.- Verification target properties.

These four full-scale verification mixes are not discarded but are cast into the previous flight set. Usually no more than two of these mixes are cast into any one casting segment.

4.3.4 Propellant Mixing

A flowchart of the SRM propellant process mixing is given in Figure 4.3-4. The first step in the mixing process occurs in the premix building. A premix consists of HB polymer, aluminum powder and iron oxide (Fe_2O_3) combined and premixed in a 600-gallon mixer bowl for 5 minutes. The Epoxy Curing Agent (ECA) is then puddled on top of the premix. ECA addition is usually held until the last minute since the propellant mix must be cast into a segment no longer than 6 hours after ECA addition.

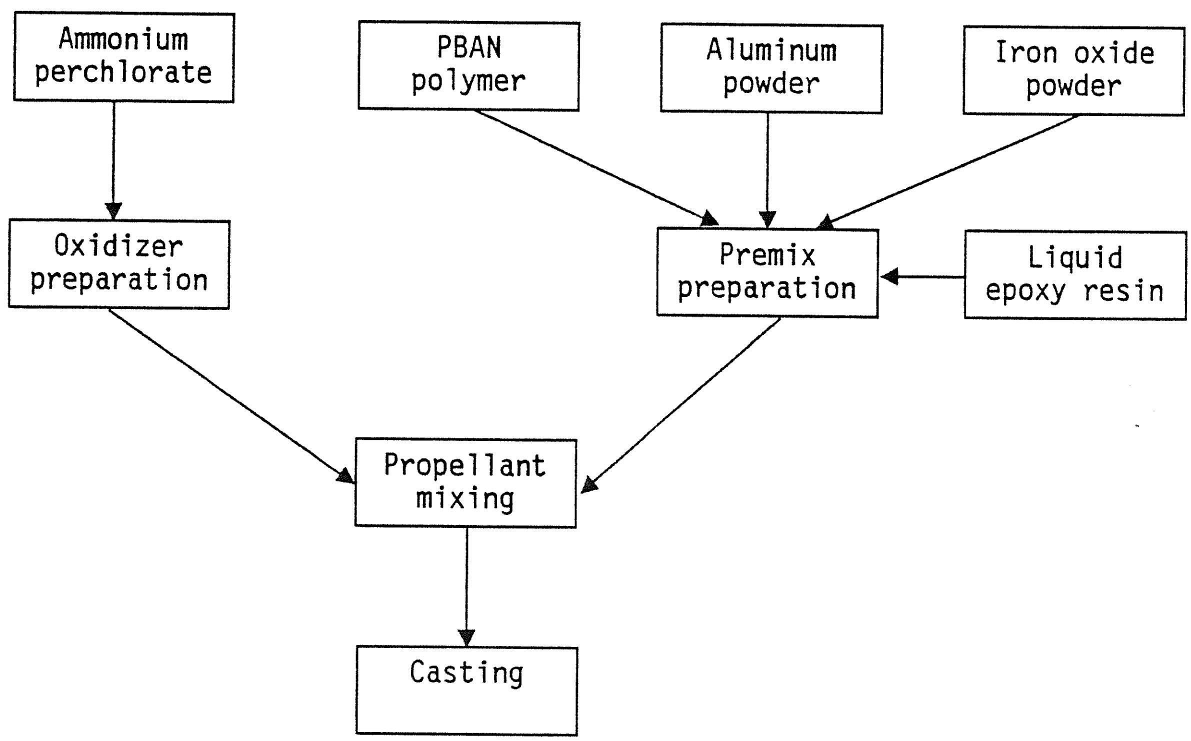


Figure 4.3-4.- SRM propellant process flow.

The premix is then taken to the mixer building where final mixing is done with the incremental addition of ground and unground AP (oxidizer). This is done with extreme care, as ammonium perchlorate is extremely unstable, decomposing with explosive violence under heat or shock. A bimodal mixture of AP is used in the propellant to give optimum propellant processing and physical property characteristics and to provide a propellant burn rate in the desired range (fig. 4.3-5).

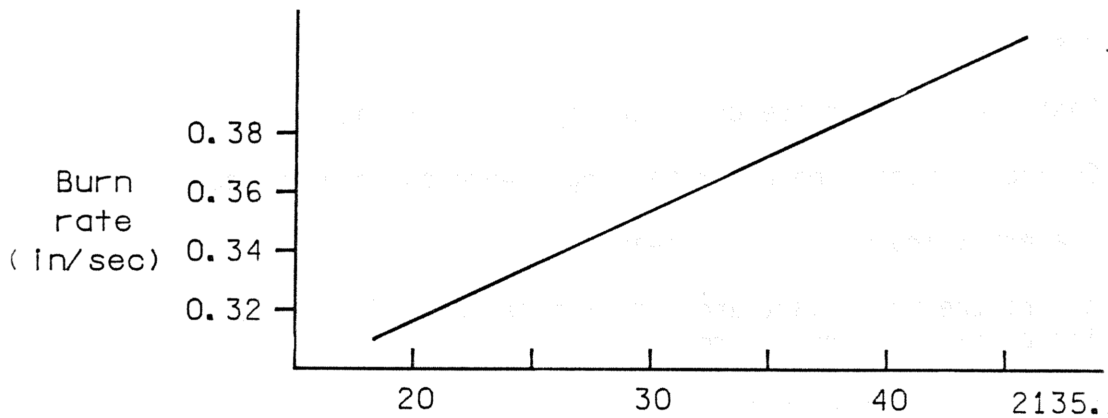


Figure 4.3-5.- Percent ground oxidizer versus burn rate.

For the SRM's a 70/30 unground/ground ratio of AP has been selected to optimize the processing properties. Fine tuning of the target burn rate is accomplished by the addition of the Fe_2O_3, as determined by the standardization/verification processes. A total mixing cycle is $\approx$50 minutes with the AP addition taking 25 minutes of that time. A minimum of 15 minutes mixing is required after completion of AP addition. The mixer bowl is then moved to the casting pit and dumped into a hopper above the segment being cast.

Before casting the propellant into a segment, the following in-process tests are made through laboratory analysis:

- Burn rate - liquid strands, six per mix
- Total solids
- HB polymer content
- Ground oxidizer particle size distribution (run on ground oxidizer only; not propellant)
- Al and Fe_2O_3 content
- Oxidizer content (AP)

All tests must be within specification before casting the mix in a segment. If the mix is not within specification it is scrapped.

4.3.5 Segment Casting

Each casting segment takes approximately 40 600-gallon mixes. Casting is accomplished by casting two identical segments sequentially to assure that the two SRM's of a flight set have burn rates that are as close as possible; i.e., cast two aft, or two center segments, etc. From the mixes that are cast into the SRM segments, the following steps are performed to verify the propellant characteristics:

- From each mix
  - Cast a 0.5-gallon ice cream carton for mechanical properties
  - Collect a pint sample for testing in-process quality control testing
- From every third mix, load a 5-inch CP

After curing the propellant and test samples at 135° F for 96 ± 4 hours, the following properties are determined:

- Mechanical properties from ice cream cartons
  - Stress - maximum 60° F - 100 psi
  - Strain at Rupture - 0° F - 40 percent, 74° F - 41 percent, 100° -38 percent
- Burn rate from 5-inch CP's

The 5-inch CP burn rate data from the mixes cast into the segments are then used in developing the flight motor performance prediction.

4.3.6 References

An Orientation - Training Course on the Redesigned Solid Rocket Motor (RSRM), TWR-16468 Rev A, Morton Thiokol, Inc. April 1987.

4.4 SRM NOZZLE

4.4.1 General

The SRM hot exhaust gases are channeled through a convergent - divergent nozzle using an ablative carbon cloth phenolic liner over steel and aluminum support structure. The nozzle is a modular-type construction with parts grouped into assemblies to facilitate maximum reuse and refurbishment of structural members. Figures 4.4-1, 4.4-2, and 4.4-3 show configurations of the SRM nozzle from different perspectives.

The SRM nozzle has gimbal capability provided by dual hydraulic rock and tilt actuators transmitting nozzle rotation about a flexible bearing. The nozzle is partially submerged in the aft SRM segment to minimize erosive conditions in the aft end of the motor and to fit within envelope length limitations.

Requirements that drove the design of the SRM nozzle to its present configuration include the following:

- Suitable SRM thrust versus time profile during ascent
- Maximum gimbal angle and gimbal rate for thrust vectoring control
- System reuse, checkout, and maintenance before and after each flight
- Capability for jettisoning the aft half of the exit cone after burnout to reduce water impact loads on reusable parts in the nozzle and for range safety thrust termination, if required
- Survivability during water impact post SRB burnout and parachute reentry. Ability to withstand a corrosive salt water environment during tow back to KSC

A breakdown of the primary materials used in the SRM nozzle components is shown in figure 4.4-4. The structural safety factor required for all components is 1.4. The erosion safety factor required for all nozzle ablatives is 2.0, excluding the last 48 inches of the exit cone, where it is 1.5. Some areas of the nozzle have safety factors less than these requirements as a result of approved waivers and deviations from MSFC and JSC. Redundant and verifiable seals exist between major assemblies as identified in figure 4.4-3 to protect against hot gas erosion and blowby.

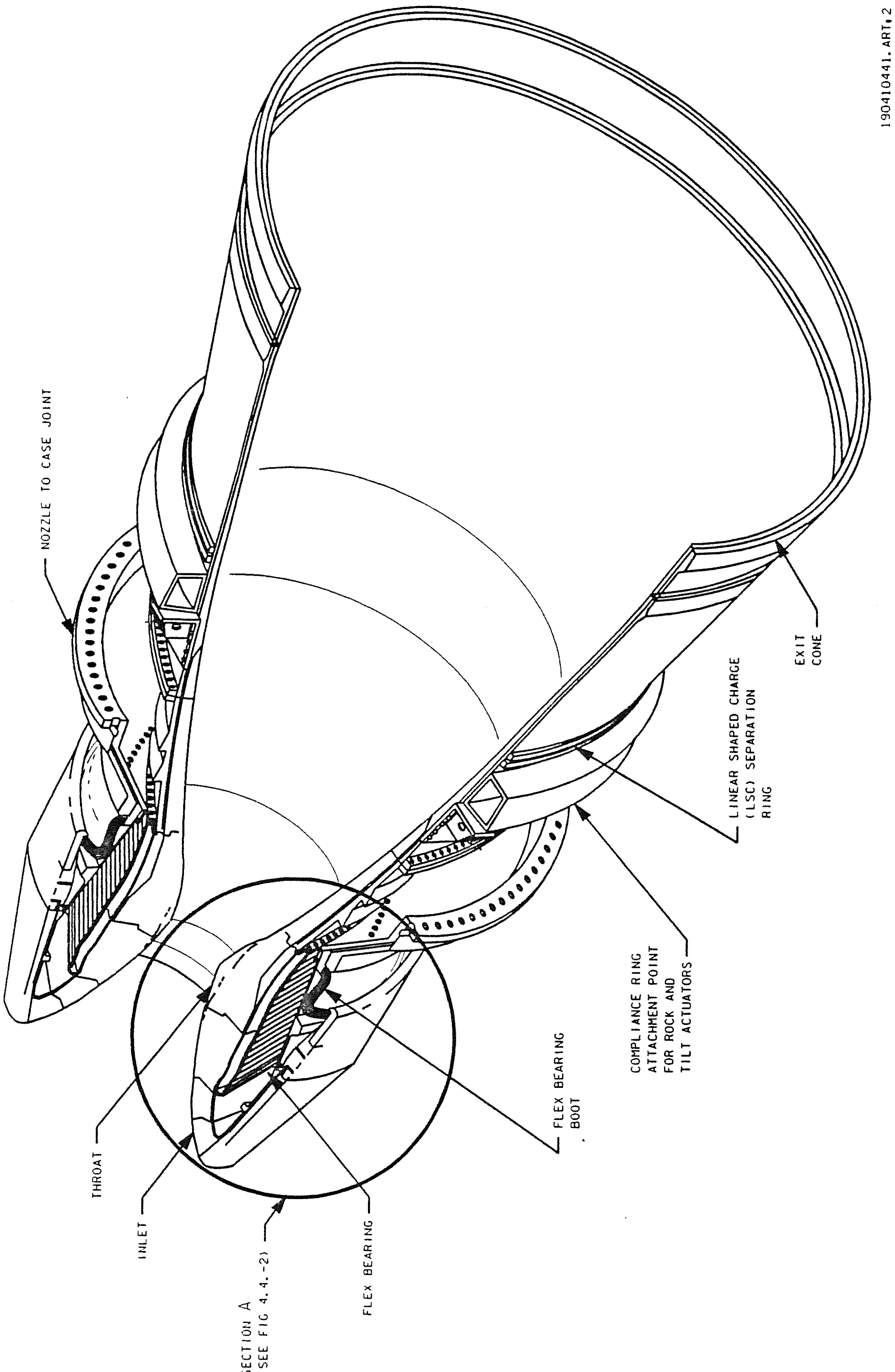


Figure 4.4-1.- SRM nozzle.

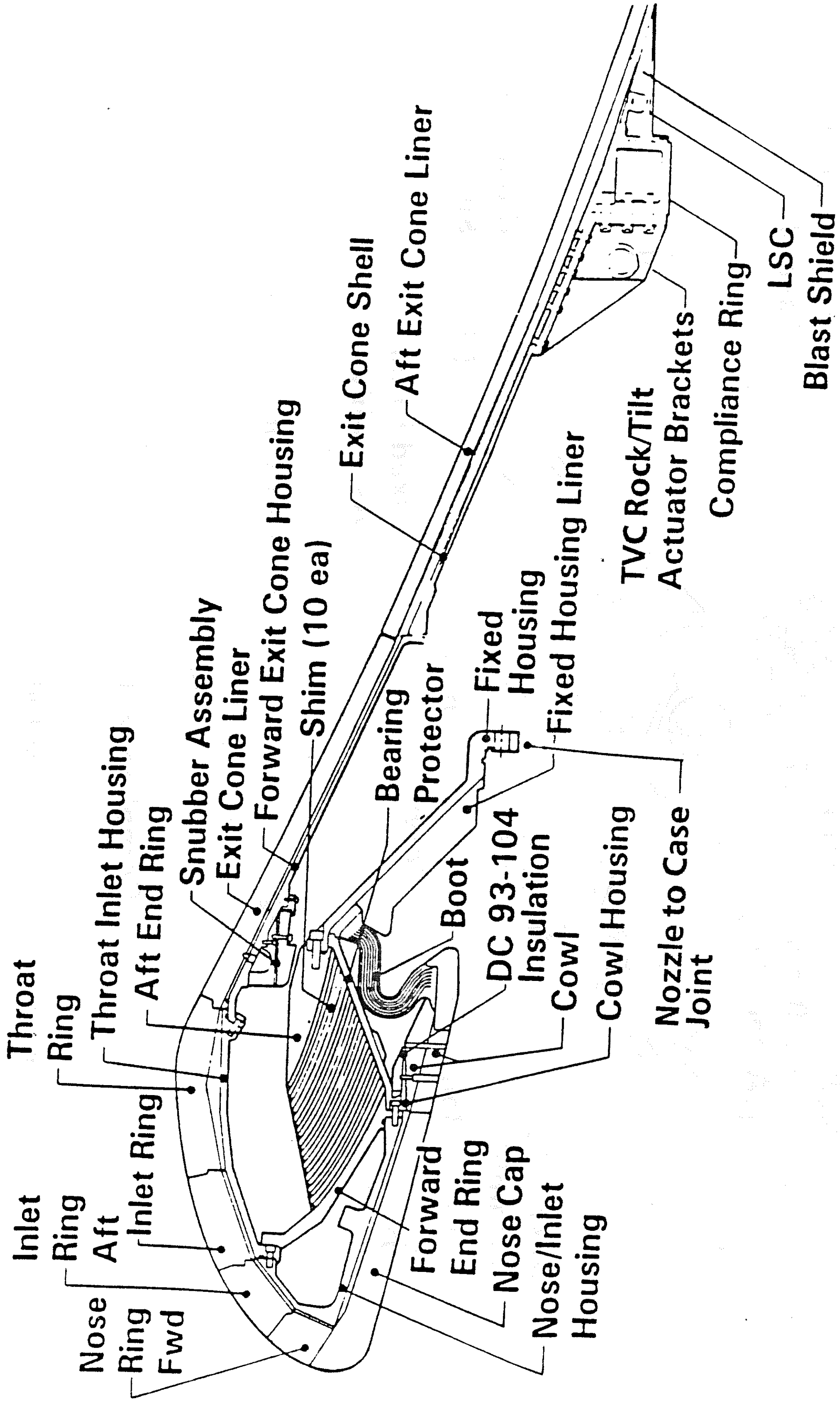
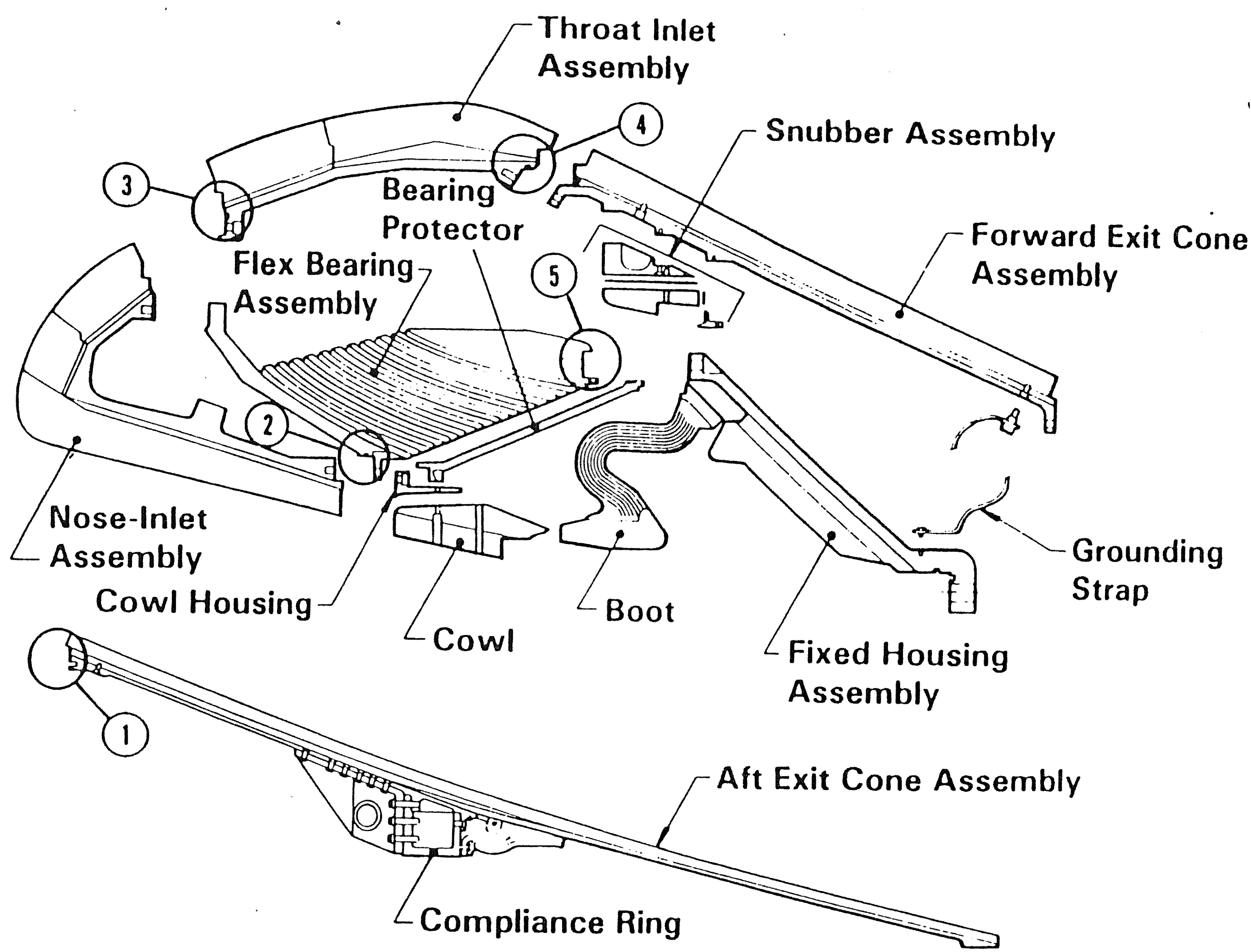


Figure 4.4-2.- SRM nozzle configuration (section A).



(1) → (5) Represent locations with redundant and verifiable seals.

Figure 4.4-3.- Nozzle internal joints.

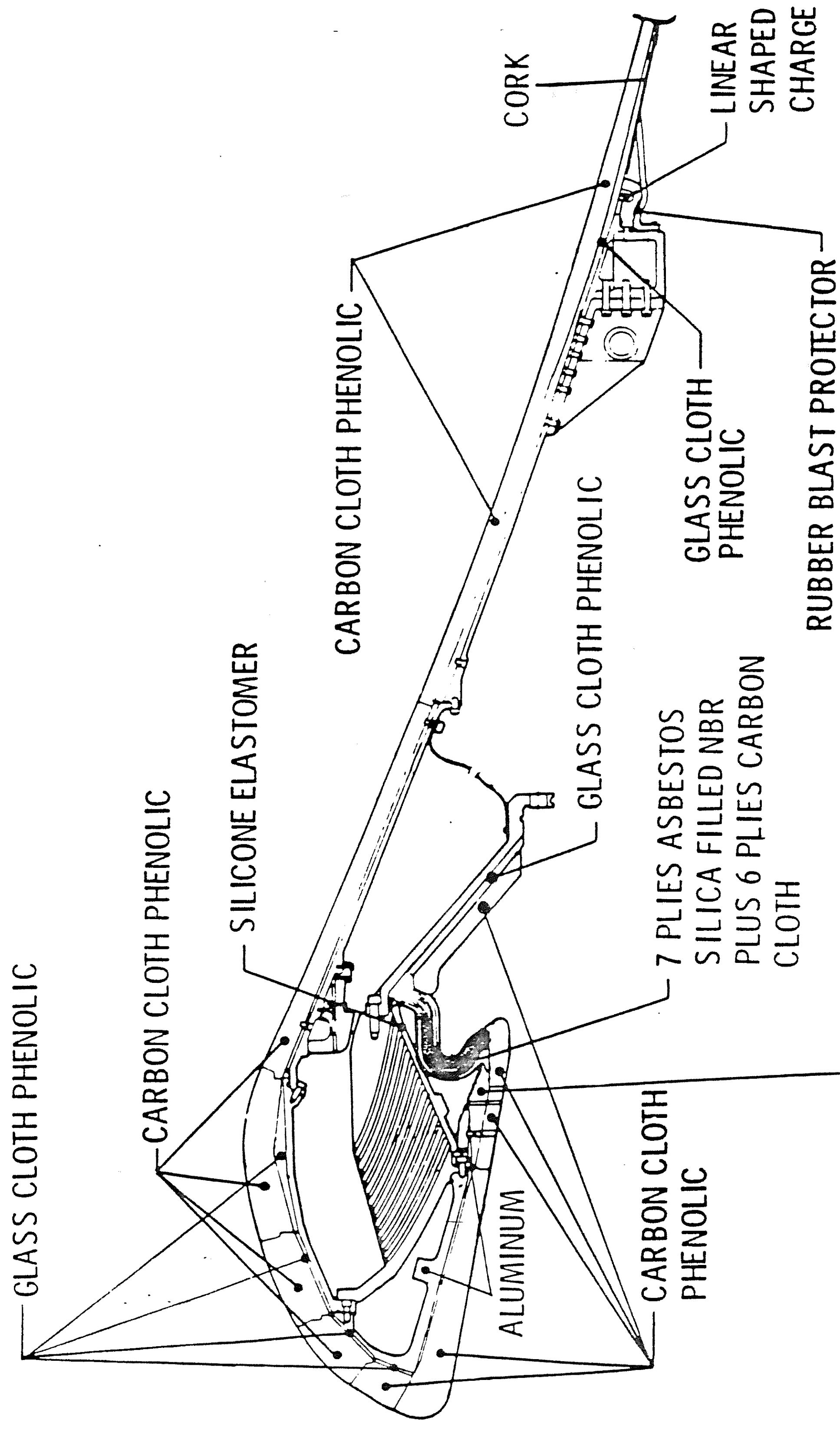


Figure 4.4-4.- SRM nozzle materials.

Additional nozzle characteristics are shown in table 4.4-I.

TABLE 4.4-I.- SRB NOZZLE CHARACTERISTICS

Nozzle type	Contoured
Expansion ratio	7.79:1
Pressurized maximum diameter (inch)	154.9
Initial throat diameter (inch)	53.9
Nozzle total length (inch)	177.7
Submergence ratio	0.226
Initial L/R_T	5.28

4.4.2 Functional Description

4.4.2.1 Cowl Assembly

The cowl assembly consists of an aluminum housing that attaches to the nose/inlet/bearing assembly and supports the outer diameter (OD) ablative liner. Refer to figure 4.4-3. The inner diameter (ID) of the cowl housing is protected by an elastomer liner. This assembly protects the boot and flex bearing from chamber radiation conditions and thermally protects the nozzle structure.

4.4.2.2 Nose/Inlet Assembly

The nose/inlet assembly interfaces with the throat assembly and bearing forward end ring. Refer to figure 4.4-3. It consists of an aluminum superstructure that is insulated and lined. The liner forms the gas flow contour around the nose. The forward nose ring traps the aft ring against forward motion. The nose cap, in turn, traps the forward nose ring to provide positive redundant retention. The assembly is sealed at each end to preclude penetration of hot, high pressure gas from the chamber into its void area.

4.4.2.3 Throat/Inlet Assembly

The throat/inlet assembly consists of a steel housing covered by ablative and insulative liners. Refer to figure 4.4-3. The assembly interfaces with the nose/inlet, the forward exit cone assembly, and the flex bearing assembly; and it is sealed at each end to preclude penetration of hot, high pressure gas. The shell is convergent to contain and support the throat and inlet rings, preclude downstream movement, and prevent ejection loads from reaching the exit cone.

4.4.2.4 Exit Cone Assembly, Forward

This assembly consists of a full-length steel structural housing and a composite liner-insulator which is bonded and pinned to the structural member for positive retention. Dual O-rings at the forward end composite structure interface preclude gas penetration along the bonded interface. Refer to figure 4.4-3.

4.4.2.5 Exit Cone Assembly, Aft

This assembly consists of composite liner, aluminum shell outer structure, compliance ring, and linear shaped charge (LSC) separation device. Refer to figures 4.4-3 and 4.4-5. The metal structure extends along the exit cone assembly to the compliance ring. The freestanding cone aft of the compliance ring is supported structurally by high strength glass-cloth tape overwrap. O-ring seals prevent potential penetration of exhaust gases along bonded or interfacing surfaces. The structural compliance ring of aluminum contains provisions for attachment of interfacing actuators. A separation ordnance ring, which is a linear-shaped charge, provides capability to jettison the aft portion of the exit cone after burnout. Figure 4.4-5 shows the attachment configuration of the SRB actuator to the aft skirt and nozzle compliance ring.

4.4.2.6 SRM Nozzle Plug

The SRM nozzle plug is a hemispherical polyurethane foam structure of an average diameter of 60.5 inches that is inserted in the nozzle just aft of the throat ring. The polyurethane foam has an outer coating of silicone rubber and primer and utilizes a circumferential polysulfide stiffener band. Room temperature vulcanizing (RTV) rubber is used around the perimeter of the nozzle plug to hold it in place.

The function of the plug is to provide the nozzle environmental protection during shipping and stacking and to protect the SRM propellant from premature ignition from SSME exhaust gas during ignition transient and pad fires. The plug is designed to be expelled upon SRM ignition without damaging the nozzle or orbiter.

4.4.2.7 Fixed Housing Assembly

The fixed housing assembly consists of a conical steel shell insulated with carbon-filled silicone rubber. It extends from the aft case to the flexible bearing aft end ring and is the primary support structure for the nozzle assembly. Refer to figure 4.4-3. It is subjected to both case pressure loads and axial compressive blowout loads. The fixed housing interfaces with the aft case boss and provides for dual O-ring seals to ensure positive sealing of this critical interface. A pressure port between the two seal locations allows pressure testing of both seals after installation of the forward nozzle assembly on the motor aft segment.

4.4.2.8 Flexible Bearing Assembly

The flexible bearing assembly provides an omnidirectional gimbal capability. Refer to figure 4.4-3. The bearing consists of a flexible core that is contained between two D6AC end rings and is thermally protected by a flexible silica-filled nitrile butadiene rubber (NBR) boot. The flexible core is a laminated structure consisting of 10 spherical D6AC shims and 11 natural rubber pads. End rings and shims absorb the applied loads while simultaneously controlling bearing motion during vectoring. Elastomeric pads transmit the loads while allowing relative motion to occur between the structural members. The inside and outside surfaces of the flexible core are protected from environmental effects by coatings of RTV rubber. The end rings have a protective coating of paint.

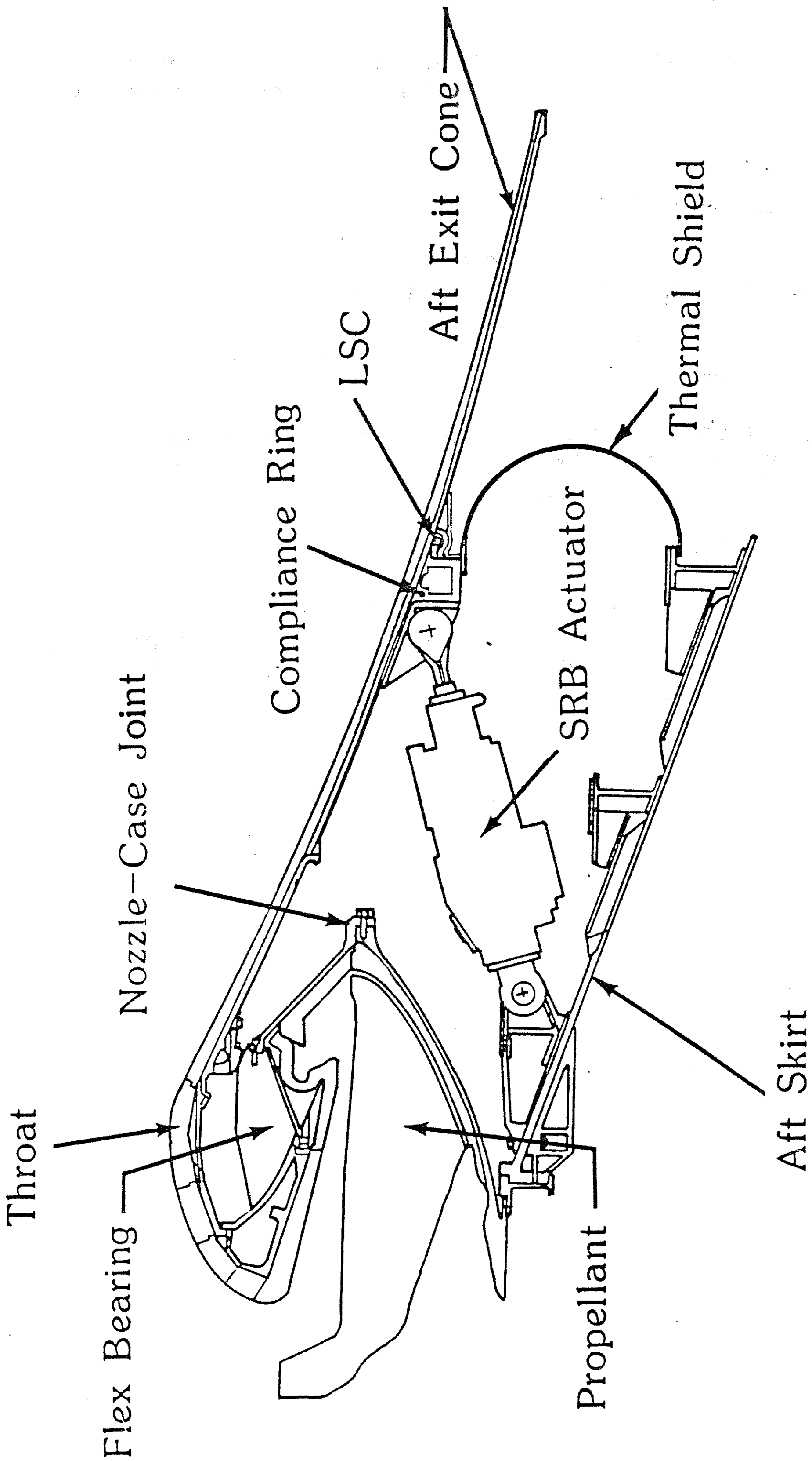


Figure 4.4-5.- Nozzle actuator design.

4.4.2.9 High Performance Motor

The main differences between the standard motor and the high performance motor (HPM) are changes in the nozzle design. These differences are shown on table 4.4-I.

4.4.3 SRM Nozzle Non-Destructive Testing

The following processes and techniques are currently used to provide non-destructive testing (NDT) of the nozzle prior to each flight:

- Metal parts - ultrasonics, magnetic particle inspection, and dye penetrant inspection
- Phenolic components - alcohol wipes for delamination indications, tag end testing, percent volatiles, compressive strength, percent resin, hardness, specific gravity, radial and tangential X-ray inspections
- Bondline evaluations - tensile button tests to verify adhesive
- Flex bearing acceptance testing/axial deflection - spring rate, tensile leak, pivot point eccentricity, torque characterization, structural verification to maximum expected operating pressure (MEOP), and stretch test for bond inspection

4.5 SRM IGNITER

4.5.1 General

4.5.1.1 Ignition System Components

The SRM ignition system is a forward end, internally-mounted, solid-rocket-type igniter. It is 43.8 inches long and contains 191 lb of propellant. The ignition system shown in figure 4.5-1 consists of the following:

- A safe and arm (S&A) device that has a reusable electromechanical actuation and monitoring assembly containing an electric motor, a manual safing and locking mechanism, a visual position indicator, and an electrical circuit switch deck. It also contains a nonreusable barrier-booster assembly containing the motor pressure seal, the safety barrier (rotated to "ARM" by the actuation and monitoring assembly), two NASA standard initiators (NSI's) with a pyrotechnical booster charge and an "armed" and a "safe" indicator electrical switch.

- An igniter adapter that provides the mounting point between the other ignition system components and the motor forward case segment.

- An igniter initiator which is a small, multinozzle, steel cased, solid propellant igniter containing a case bonded 30-point star propellant grain.

- A rocket motor igniter which is a single nozzle, steel cased, internally and externally insulated solid propellant igniter containing a base bonded 40-point star propellant grain.

- Propellant in the igniter and initiator is identical and they are cast at the same time from the same batch of propellant. The propellant consists of 69 percent ammonium perchlorate, 10 percent spherical aluminum, 3 percent ferric oxide, and 18 percent P BAN polymer and epoxy resin.

Improvements in the igniter design were made with the introduction of a thicker adapter plate beginning with STS-50. Further improvements are scheduled with STS-59 to include a bonded igniter J-seal to replace zinc chromate putty. These changes will reduce the liklihood of hot gas erosion and blowby to the igniter gasket seal region.

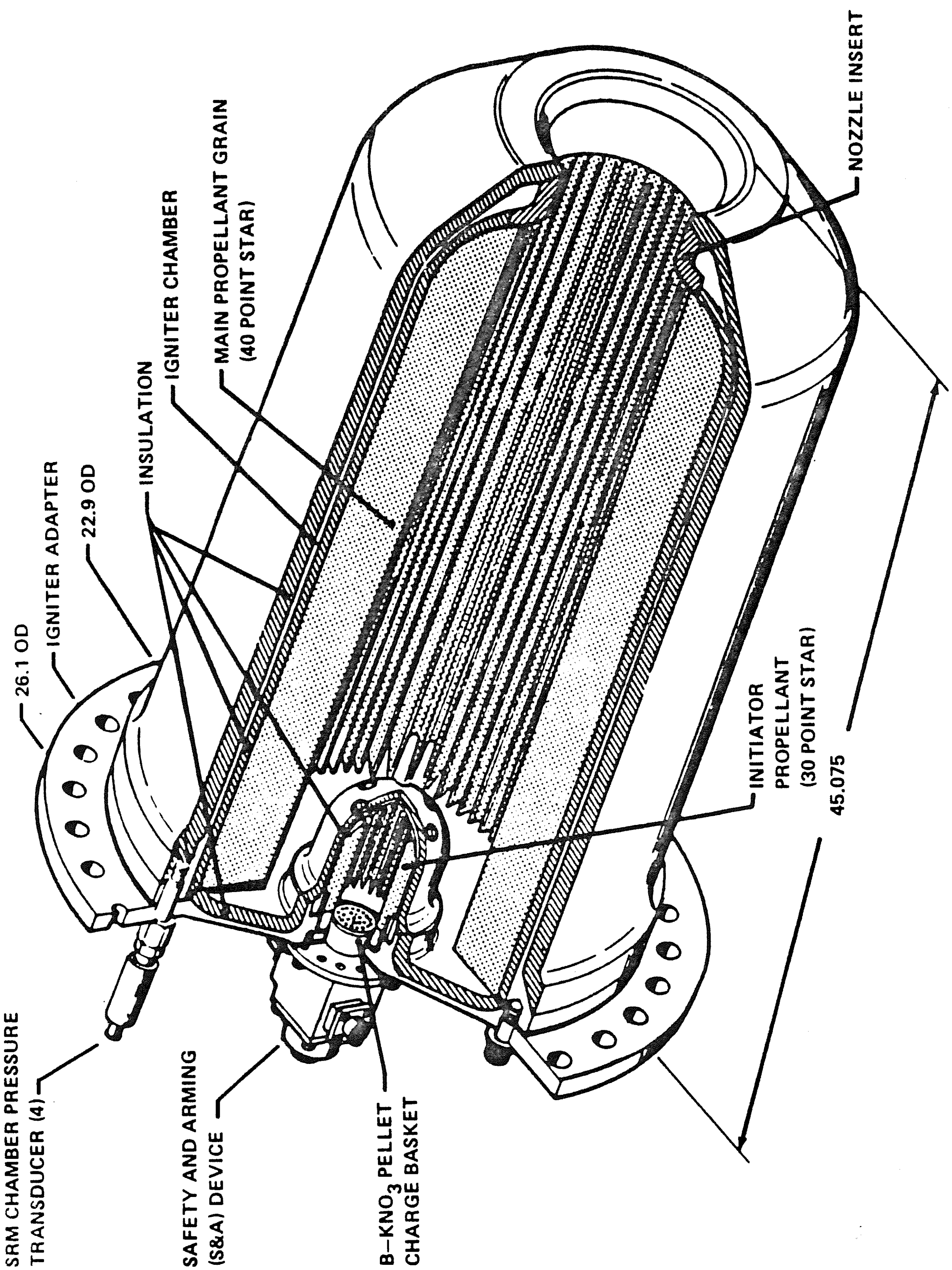


Figure 4.5-1.- SRM igniter.

4.5.1.2 Ignition Design Characteristics

For the ignition sequence, the manual lock pin must be removed from the S&A device; an electrical arming signal to the S&A device causes the barrier rotor to move into the armed position. In the armed position, the NSI's fire through a thin barrier seal into the pyrotechnical pellet charge which is retained in the S&A device behind a perforated plate. The pellet charge ignites the propellant of the igniter initiator whose combustion products in turn ignite the propellant of the rocket motor igniter. A profile of the SRM initiator, ignitor, and motor chamber pressure as a function of time is shown on figure 4.5-2. The igniter initiator is threaded directly into the igniter adapter with thread sealing compound, whereas the S&A device is bolted to the adapter with redundant, verifiable seals. Both the igniter initiator and adapter are insulated to provide protection from the heat of the propellant gases. The adapter is reusable and has heavy insulation to ensure structural and dimensional integrity for 20 uses. The igniter initiator is not reusable. The steel igniter chamber is bolted to the adapter with redundant, verifiable seals and is insulated both externally and internally. The insulation is provided to ensure a low temperature profile in the igniter chamber during motor operation and SRM descent and is primarily required because of radiant heating and recirculation of motor gases through the large axial, aft throat of the igniter. The silica-phenolic throat insert has a thick profile to ensure reusability of the igniter chamber without degradation because of excessive heating. The igniter adapter is bolted to the forward motor segment with redundant, verifiable seals.

4.5.2 Functional Description

4.5.2.1 S&A Device

The device shown in figure 4.5-3 consists of an actuation and monitoring (A&M) assembly containing an electromotive drive element, electrical switch, and manual mechanical safing features that are designed to be reused a total of 20 times and a pyrotechnic barrier-booster (B-B) assembly that contains provisions for mounting two NSI's, a mechanical safe-arm barrier, and a pyrotechnic booster assembly. The B-B assembly is shown in figure 4.5-4.

4.5.2.2 Igniter Adapter

The igniter adapter is a machined steel forging, internally insulated with silica-asbestos filled NBR. The adapter is mounted on the SRM forward polar boss with forty 0.75-inch-diameter, high-strength steel bolts. "Gask-O-Seals" are used on the bolt circle to provide redundant seals with leak test provisions. Four special bolts are provided for mounting the pressure transducers to measure the motor chamber pressure. One machined port is provided for installation of the igniter pressure transducer. Two 0.25-inch tube ports are also provided for leak testing the igniter-to-adapter and adapter-to-motor seals. Provision is also made in the S&A device to leak

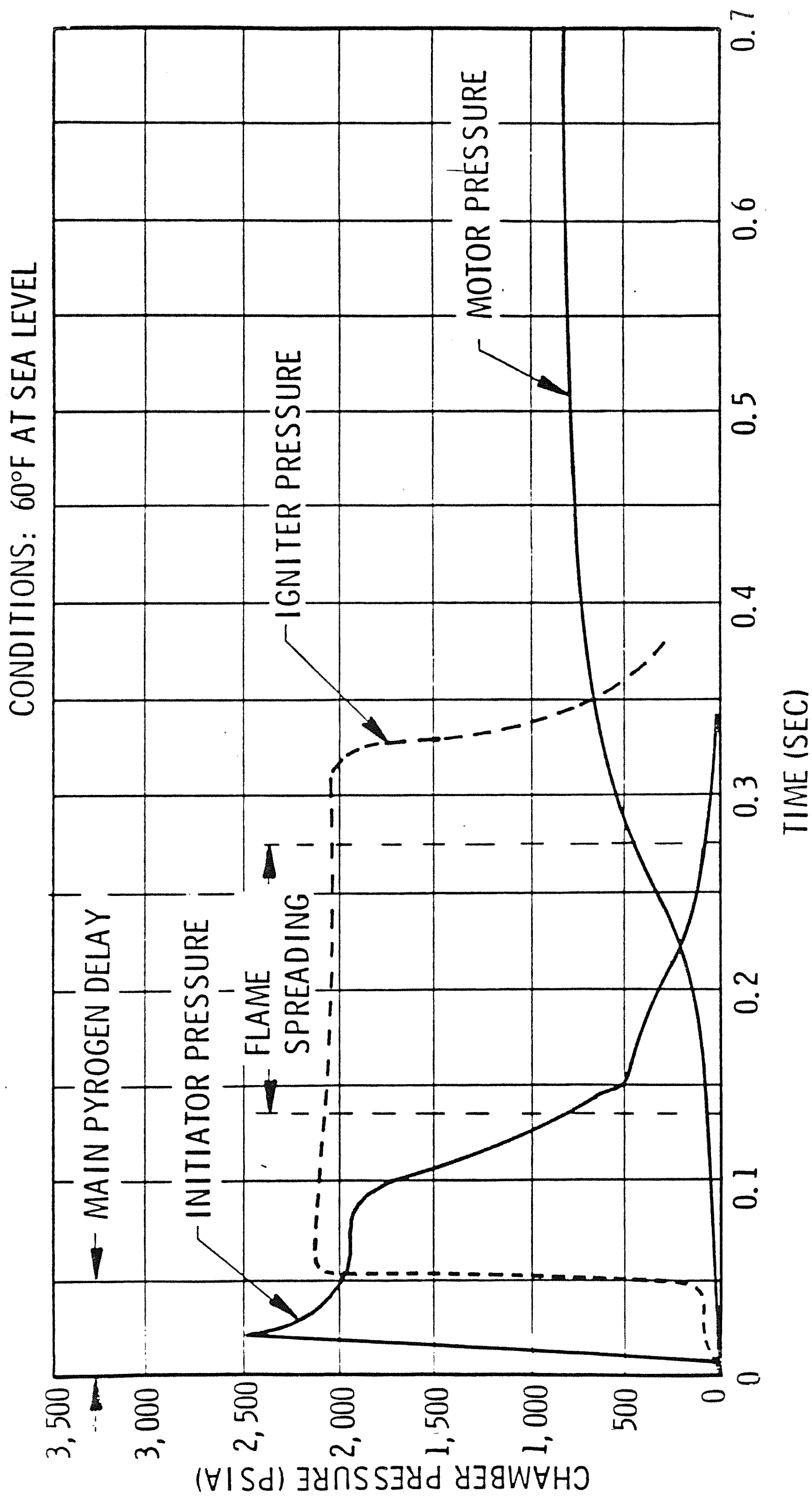


Figure 4.5-2.- SRM ignition sequence.

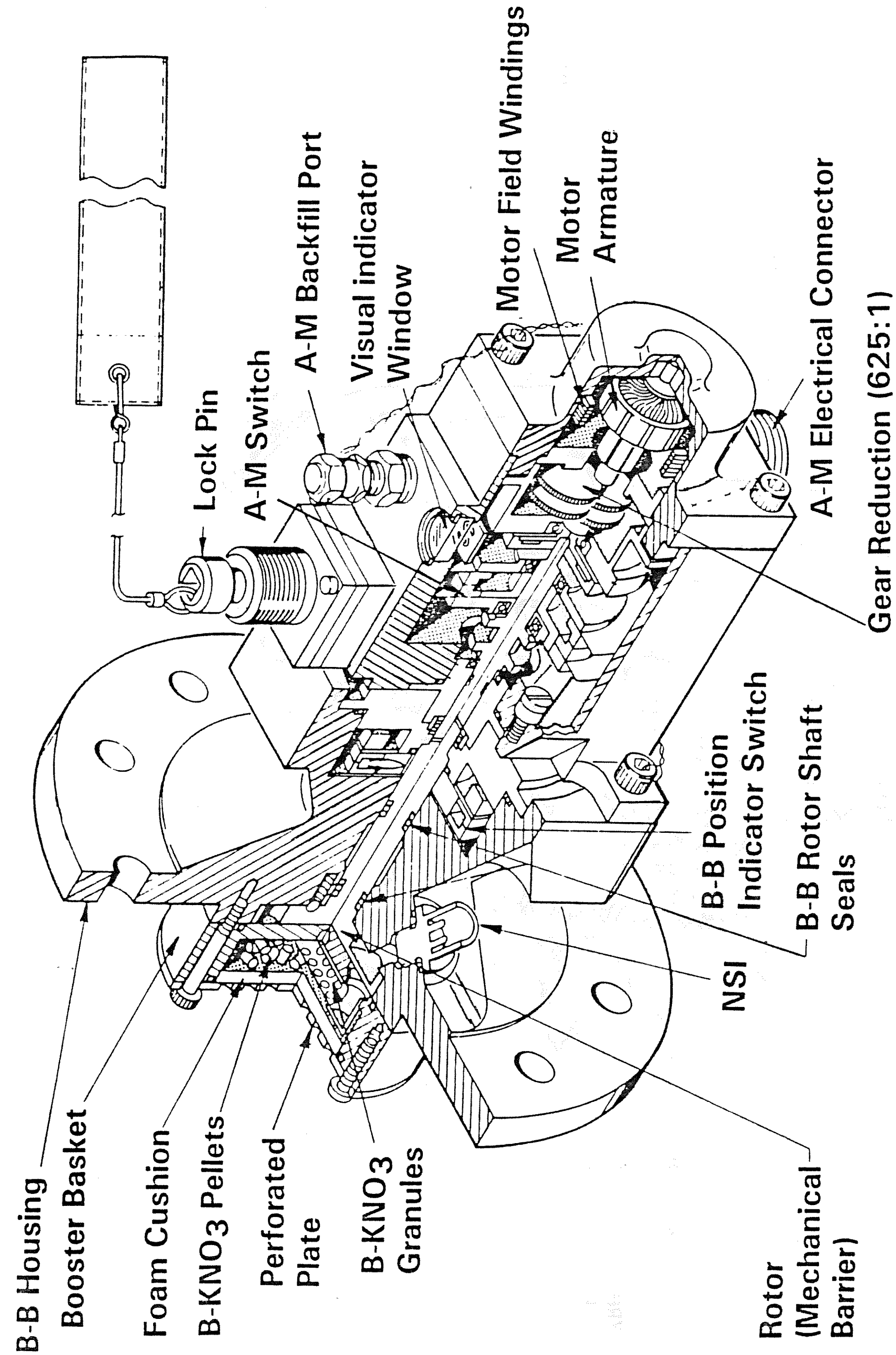


Figure 4.5-3.- SRB SRM ignition S&A device.

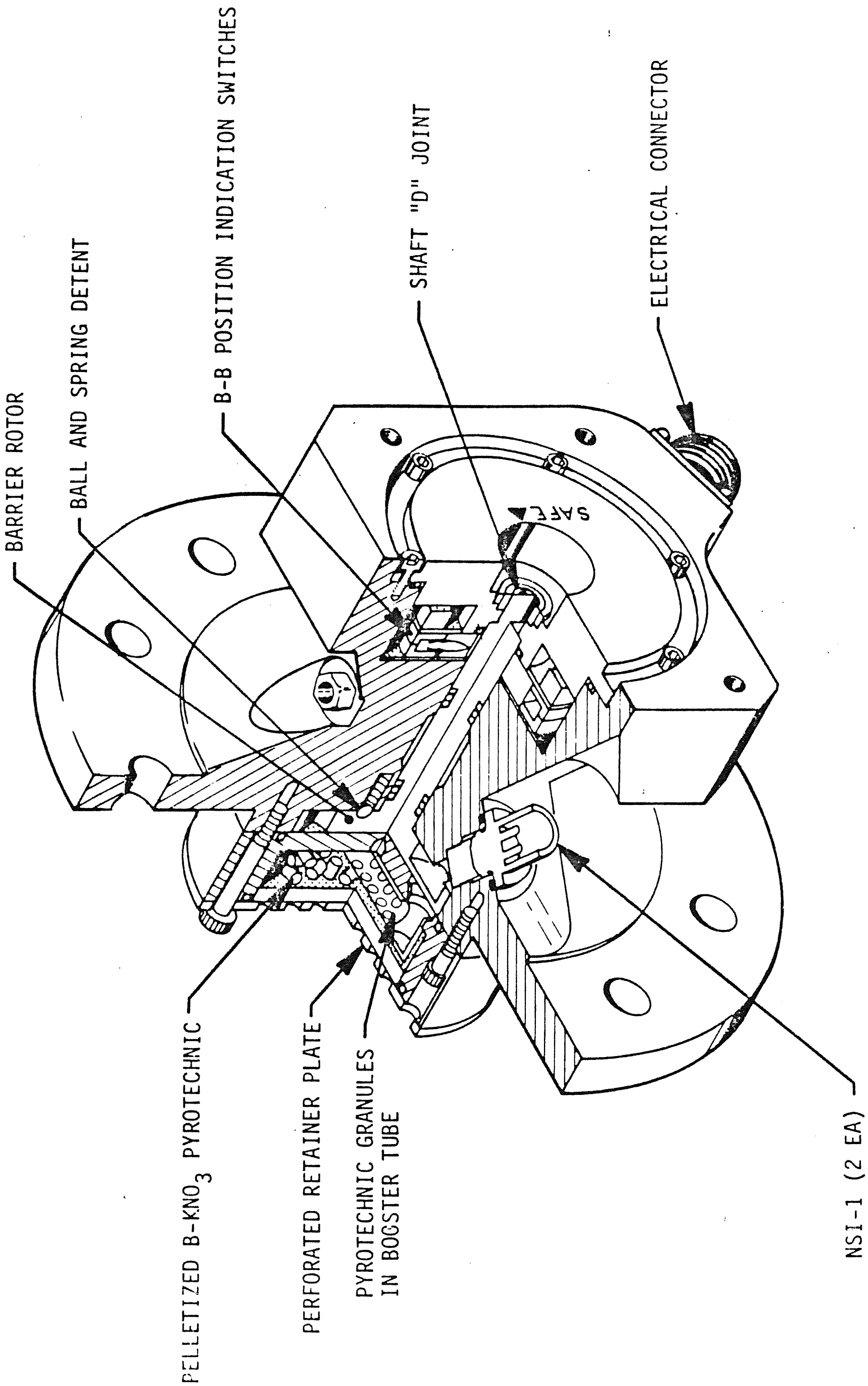


Figure 4.5-4.- Modified barrier booster assembly.

check the unit. See figure 4.5-5 for a comparison of the present igniter adapter design and the STS-59 and subs design.

4.5.2.3 Igniter Initiator

The igniter initiator is basically a small multinozzle rocket motor. The igniter for the igniter initiator is the pellet charge in the S&A device. In order to provide a simple attachment, this component is threaded directly into the adapter. The steel case is externally insulated with silica-asbestos filled NBR which is premolded and secondarily bonded in place with adhesive. This adhesive is also used to bond in the carbon-phenolic throat inserts and the silica-asbestos filled NBR internal insulation in the forward end. The entire case is lined with 0.05-inch-thick liner. A case bonded grain of propellant is cast into the lined case in a 30-point star configuration.

4.5.2.4 Rocket Motor Igniter

The rocket motor igniter consists of a steel chamber bolted to the igniter adapter. In order to obtain the desired igniter ballistics, the chamber ID is 17.8 inches with 0.35 to 0.65-inch of silica-asbestos-filled NBR internal insulation. Since the forward motor case opening is fixed at 20.97 inches, it is necessary to bolt the igniter to the adapter using external bolts to maintain positive structural margins for the igniter chamber flange for 20 uses. Thirty-two 0.75-inch, high-strength steel bolts are used to attach the chamber to the adapter. Redundant seals are provided by using Gask-O-Seals inside and outside the bolt circle and Stat-O-Seals under each bolt. Leak test provision is made in the adapter to ensure that there is no leakage in any of the seals after assembly. The igniter case is insulated externally with 0.98-inch of silica-asbestos filled NBR to protect it from motor gas temperatures both during SRM firing and subsequent heat soak during descent. The silica-phenolic throat insert is also designed to provide similar protection. The internal insulation of the igniter is coated with a 0.05-inch-thick layer of liner to ensure adhesion between the propellant grain and the internal insulation. The case bonded propellant consists of a 40-point star grain of propellant having an initial surface area of 8410 sq^2. When ignited at 60° F, this grain produces a nominal operating pressure of 2020 psia and nominal mass flow of 550 lb/sec for 0.270 seconds. The 7.200-inch ID igniter throat consists of a machined silica-phenolic insert with a 0.030-inch shear ply of silica-asbestos filled NBR between the steel case and the throat. The silica-phenolic cloth layers of the throat insert are oriented to provide the optimum strength of the part to prevent any possibility of ejection.

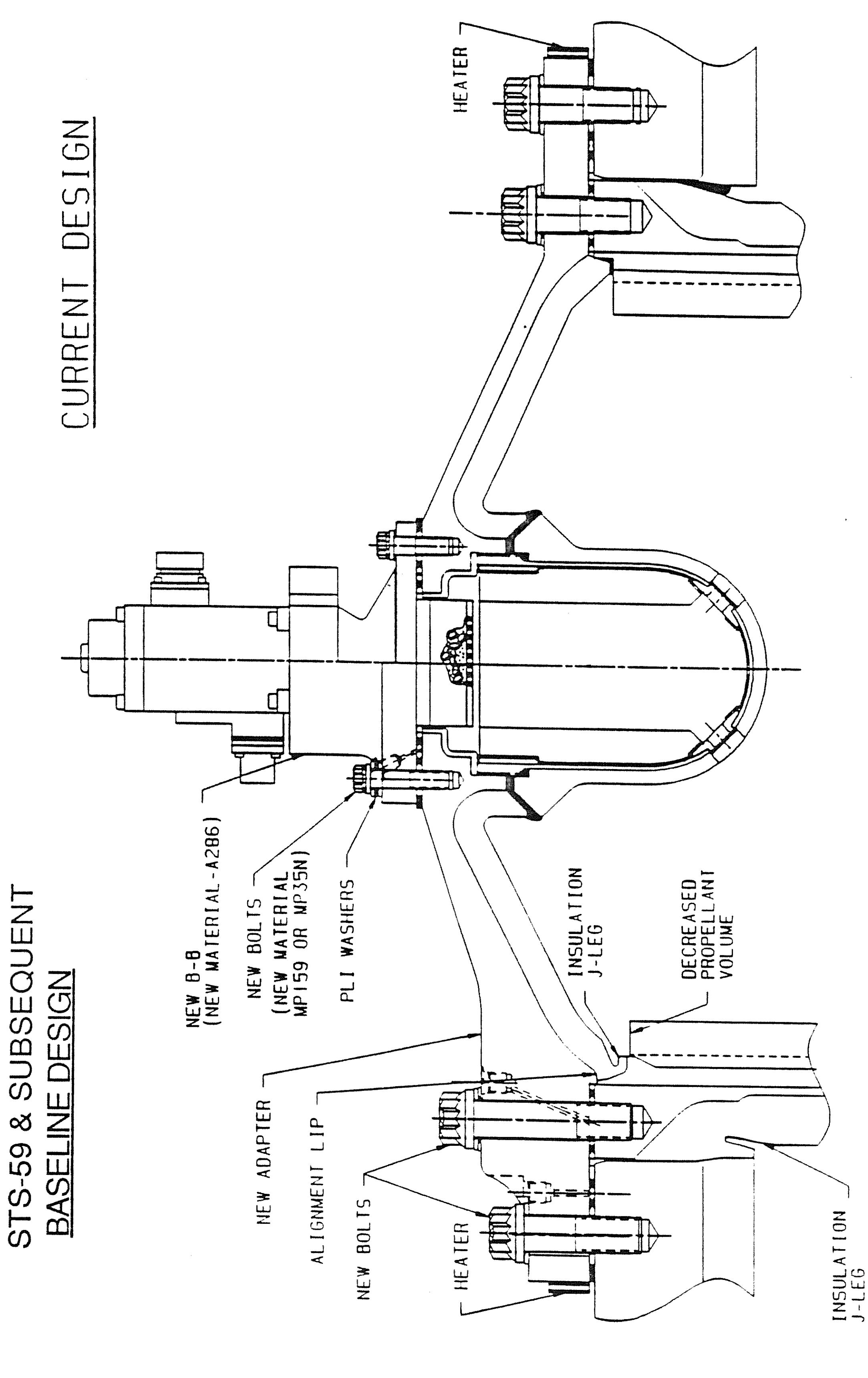


Figure 4.5-5.- RSRM igniter redesign.

4.6 SRB HYDRAULIC POWER UNIT AND TVC SYSTEM

4.6.1 Introduction

The SRB thrust vector control (TVC) system is located on the aft skirt and consists of two hydraulic power units (HPU's) and two servoactuators. The HPU's supply the hydraulic pressure which drives the servoactuators to gimbal the solid rocket motor (SRM) nozzles in response to the TVC steering commands. This action provides ascent control authority in roll, pitch, and yaw. The two auxiliary power units (APU's) provide power for the hydraulic pumps.

The actuators are oriented at 45° outboard to the vehicle pitch and yaw axes; therefore, the SRB axes are called tilt and rock. Each HPU is connected to both actuators. One HPU serves as the primary hydraulics on an actuator; the other HPU is the secondary hydraulics on that actuator. HPU A is the primary hydraulics for the rock actuator, and HPU B is the primary hydraulics for the tilt actuator. If one HPU fails, the surviving HPU can increase its hydraulic pressure output and power both actuators at a slightly degraded nozzle gimbal velocity. The nominal nozzle gimbal rate is 5° per second; the nozzle gimbal rate with one HPU failed is 3° per second. The gimbal capability in both axes is $\pm 5°$.

The SRB TVC system is similar to the orbiter TVC system. Table 4.6-I shows some of the basic differences.

The components of the SRB TVC system will be described in this systems brief in detail. Figures 4.6-1 through 4.6-3 show an overview of the SRB HPU. The SRB systems handbook drawing 10.2 should be referred to when studying this systems brief in detail. The zone codes in the following pages refer to drawing 10.2.

4.6.2 Auxiliary Power Unit LZ3

The purpose of the APU is to convert chemical energy stored in liquid hydrazine (N_2H_4), into mechanical shaft power to drive the hydraulic pump. Each APU contains the following components:

A. Fuel supply module (FSM)

B. Fuel tank isolation valves and feedlines

C. APU assembly

- Fuel pump
- Fuel control valves
- Gas generator
- Gas generator heaters
- Dual pass, reentry turbine
- Gearbox

D. APU controller

Filtered, liquid hydrazine at an initial pressure of 350 to 415 psia is supplied to the fuel pump inlet from the FSM. During APU startup, the pressurized fuel flows through the isolation valve to the start bypass line and directly into the gas generator. As the turbine speed increases and the integral fuel pump output pressure exceeds the FSM supply pressure, the fuel pump begins to deliver N_2H_4 to the gas generator. At nominal APU operating speed, the fuel pump output is 1270 psi. The fuel is decomposed over a catalytic bed in the gas generator into a hot gas. The hot gas enters the turbine through the primary turbine inlet for the first pass and then re-enters the turbine through the secondary inlets for the second pass. The expanded gas is then dumped overboard through the turbine exhaust ducting. Under normal operation (100 percent) the turbine operates at a constant nominal output of 72000 rpm. When in backup mode, the turbine operates at 110 percent of nominal which is 79,200 rpm. A second backup mode operates the turbine at 112 percent (80,640 rpm). The turbine is linked to the hydraulic pump through a fixed ratio (18.93:1) gearbox that reduces turbine speed to that required for hydraulic pump operation. Other reduction gear trains in the gearbox power the fuel pump and lubrication pump.

4.6.2.1 Fuel Supply Module S Z 2

The FSM stores the liquid N_2H_4. The tank has a volume of 1950 in^3 and stores approximately 32 lbs of N_2H_4. The FSM is pressurized to about 400 psi with 1.1 pounds of GN_2 which delivers the N_2H_4 to the APU fuel pump inlet. The tank contains a diaphragm to separate the pressurant from the fuel. Temperature and pressure sensors are mounted on the tank.

4.6.2.2 Fuel Tank Isolation Valves and Feedlines R Z 2

The fuel feedline from the FSM has a 25µ fuel filter at the FSM exit, a fuel isolation valve, a pressure relief valve set at 50 psid, and a 25µ fuel filter between the fuel pump and gas generator control valves.

The fuel isolation valve (FIV) ensures positive isolation of the fuel in the FSM from the APU assembly during nonoperational periods. This normally is a closed valve. The FIV is opened at APU startup and will remain open until power is removed as part of the HPU power deadfacing at SRB separation.

The pressure relief valve allows reverse flow to the FSM in the event of fuel expansion due to heat soakback.

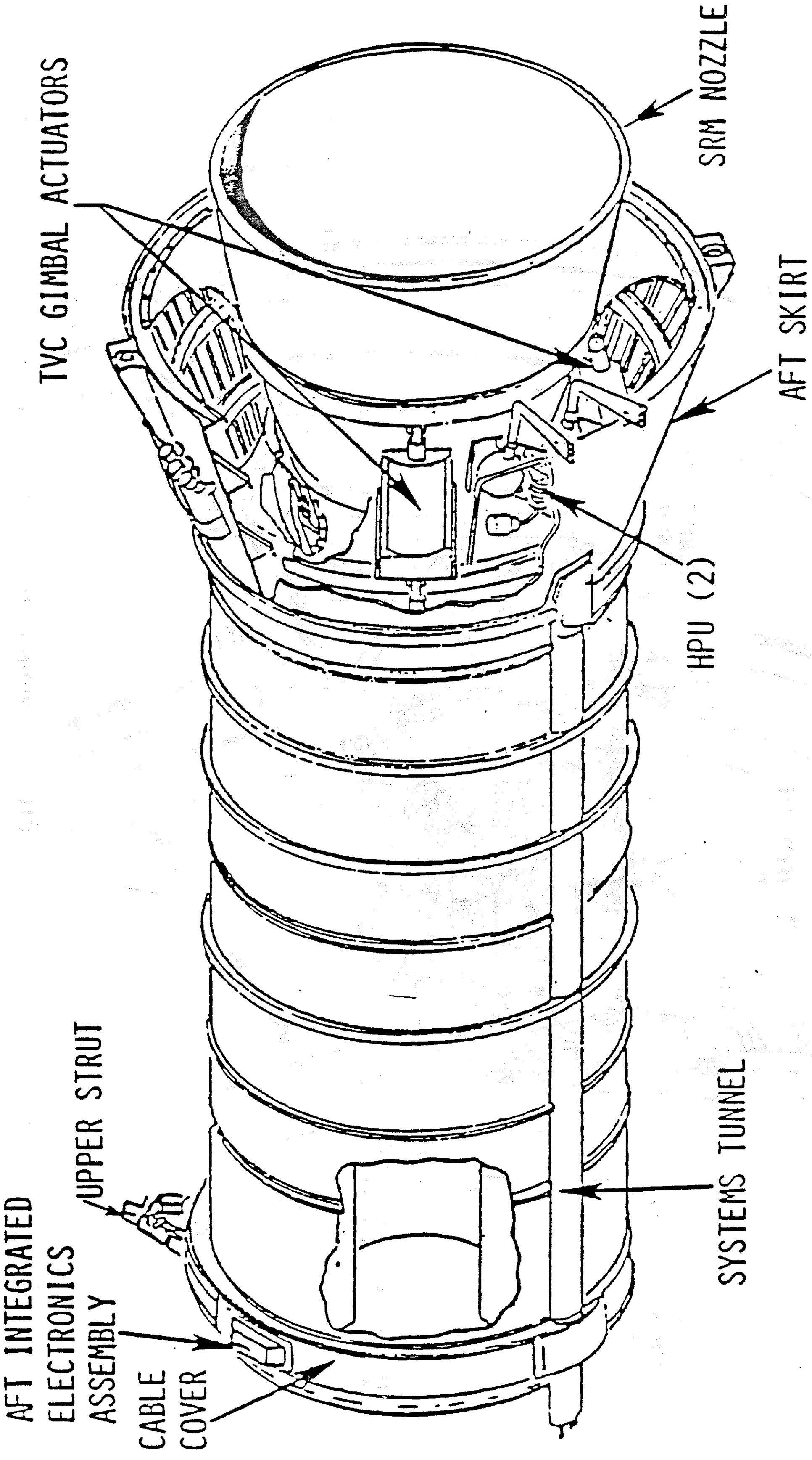


Figure 4.6-1.- SRB TVC components location.

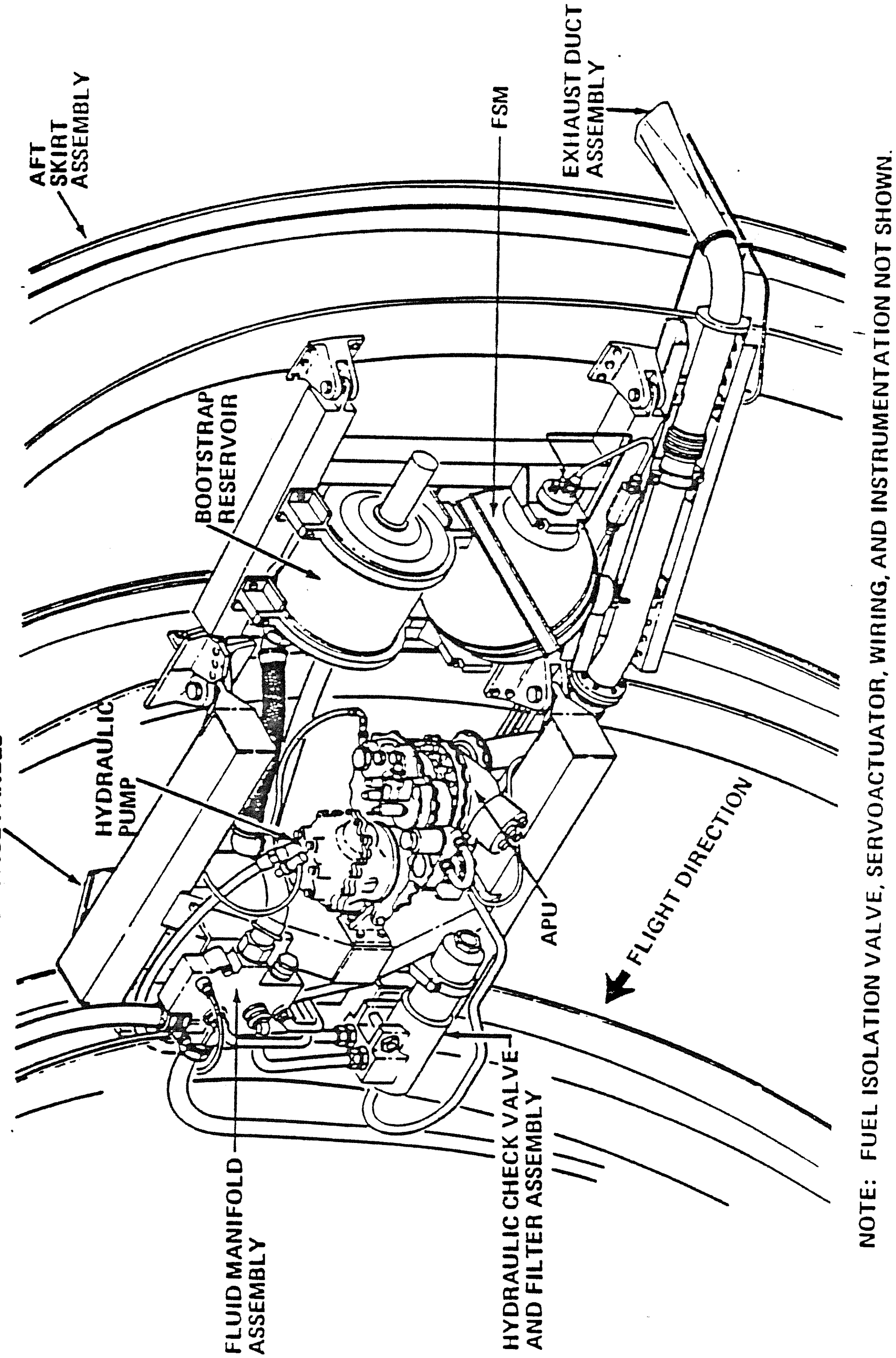


Figure 4.6-2.- SRB TVC panel locations.

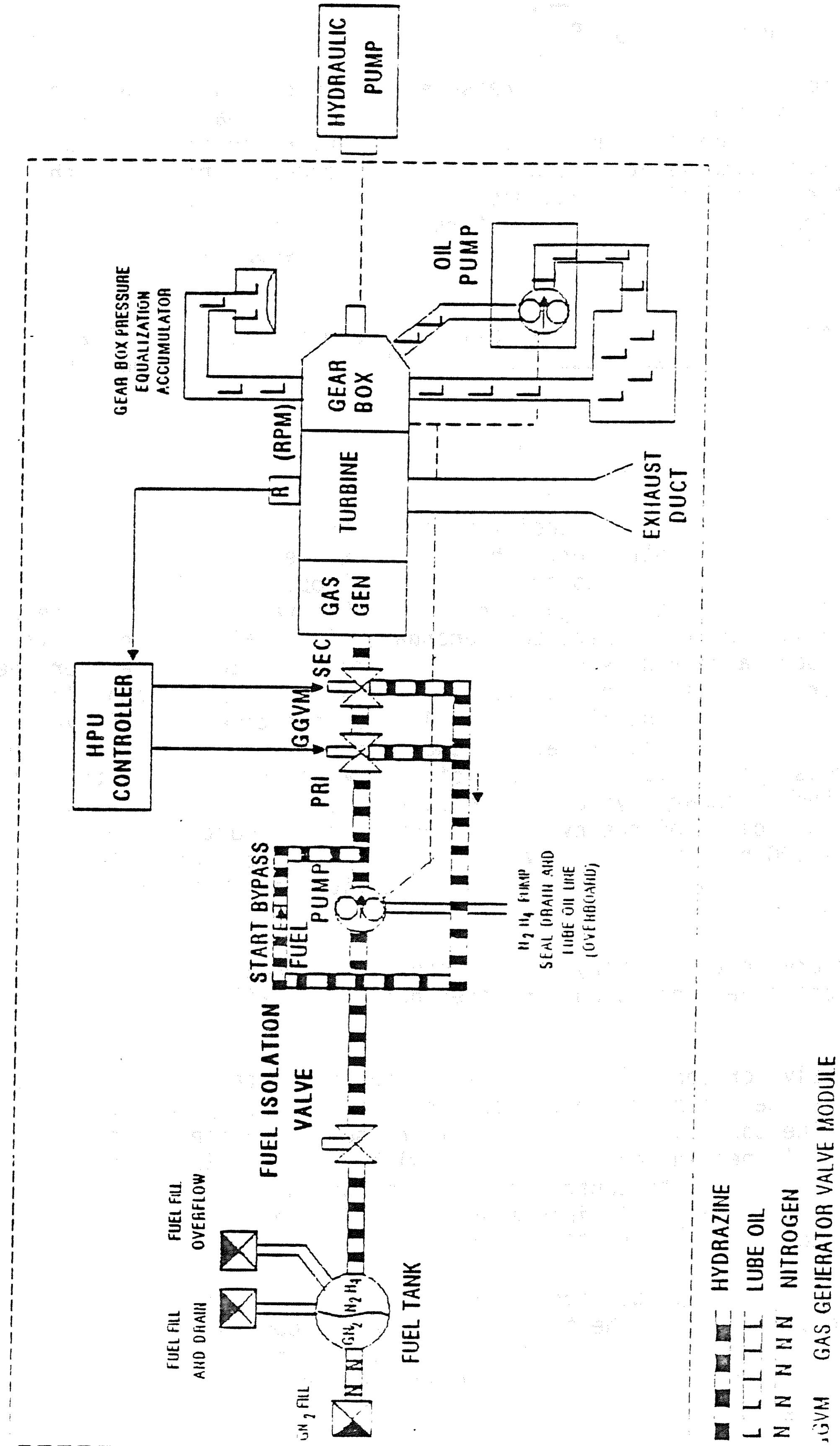


Figure 4.6-3.- HPU system schematic.

4.6.2.3 Fuel Pump Assembly PZ2

The fuel pump maintains the fuel pressure at 1270 psi to the gas generator once the APU is started. Each assembly has a start bypass valve which allows fuel to bypass the pump to the gas generator for initial APU start before the fuel pump is activated. As the APU comes up to speed, the pressure output of the fuel pump becomes greater than the initial FSM pre-pressurization pressure. Then the startup bypass line check valve closes. For the remainder of APU operation, the fuel pump supplies all of the fuel to the APU.

The fuel pump assembly also has a relief valve which is set at 1725 ± 25 psid to relieve fuel overpressure between the fuel pump and the gas generator.

4.6.2.4 Fuel Control Valves 0Z2

The two control valves are solenoid-operated, three-way, two-position valves controlled by an APU controller. The controller regulates the APU turbine speed by the operation of a control valve (full open or full closed) to regulate the fuel flow to the gas generator. One valve is called the primary (PRI) valve and the other, the secondary (SEC) valve. The PRI valve is a normally open valve and is controlled by either the 100 percent or the 110 percent control circuit. Nominally, the PRI valve is opened and closed by the 100 percent speed control circuit. At the same time, the SEC valve, which is a normally closed valve, is powered open. If the primary hydraulic pressure drops <u>between</u> 1900 and 2200 psig, a switching valve in the actuator will allow the secondary hydraulics to power the actuator. A signal is sent to the APU controller of the hydraulic system now in control. This signal inhibits the 100 percent circuit and enables the 110 percent circuit. This increased APU speed provides extra hydraulic flow needed to drive two actuators instead of one.

If the pressure in the primary system returns to normal, the HPU can revert to 100 percent speed operation, and then both HPU's will resume powering the TVC system.

If the PRI valve or control logic fails to the open state, the valve can no longer be operated to maintain the turbine speed at 100 percent. Instead, the APU turbine speed will continue to increase. When the speed reaches 112 percent, the 112 percent control circuit will close the SEC valve. Then the SEC valve will be used to control the turbine speed at 112 percent. Once the 112 percent operation is initiated, the affected APU will operate at 112 percent for the duration of SRB powered flight.

There is no overspeed protection if the control logic for the SEC valve fails to the open state. The turbine speed would continue to increase until blade failure occurs. The SRB APU turbines are surrounded by a containment ring, but this ring will only contain blade failures if the blades fail below 126 percent nominal speed.

4.6.2.5 Gas Generator M Z 2

The gas generator consists of a catalytic bed with a granular catalyst (Shell X-405) within a pressure chamber mounted inside an exhaust chamber. Hydrazine propellant decomposes when it contacts the catalyst and hot gases are produced. The hot gases are directed to the turbine. Gas generator heaters are provided to preheat the catalyst bed for smooth decomposition. Only one heater will be used at any one time and will be powered off prior to launch. The design operating gas generator temperature is 1700° F.

4.6.2.6 Dual-pass, Reentry Turbine L Z 1

The hot gas from the gas generator is routed to the turbine. The turbine converts thermal energy from the hot gas into mechanical energy. The hot gas enters the turbine through the primary inlet and is cycled to reenter the turbine secondary inlet before being exhausted overboard.

The normal operating speed is 100 percent (72,000 ± 5760 RPM). The backup speeds are 110 percent (79,200 ± 5760 rpm) and 112 percent (80,640 ± 5760 rpm). The turbines are designed to reach their nominal speed within a maximum of 4 seconds after initial fuel flow.

4.6.2.7 Gearbox I Z 2

The gearbox consists of a cluster of speed reduction gears and a lubrication system. The gearbox provides shaft power to drive the gearbox lube oil pump, APU fuel pump, and HPU pump. The speed ratios are as follows:

- turbine to HYD pump 18.93:1
- turbine to lube pump 6.07:1
- turbine to fuel pump 18.93:1

The nominal rated shaft output is 135 hp. Lubrication of the gearbox is provided by an internal oil pump. The oil returns through a filter, then to an accumulator, which is used to compensate for thermal expansion and leakage.

4.6.2.8 APU Controller J Z 6

There is one controller for each APU. A controller provides speed control, logic for APU startup and shutdown, and signal conditioning. The controller electronics are located in the aft integrated electronics assembly (IEA).

There are two magnetic pickups (MPU) on the APU drive shafts that are used to sense APU turbine speed. The output from MPU 2 is sent to the 100- and 110-percent circuits, and the output of the other MPU 1 is sent to the 112-percent circuit. The output is a train of pulses which have a frequency that is directly proportional to turbine speed. This train of pulses is changed to a rectangular wave by the waveshaping circuit. The shaped pulses are then applied to a one-shot multivibrator. The output of the multivibrator is filtered and then compared with a reference voltage in the comparator. The signal from the comparator is amplified and sent to the valve.

The 110-percent speed circuit is enabled when the pressure indication of a pressure O.K. pressure switch is lost. When the APU's are first started, the 110-percent speed circuit will be enabled until the hydraulic pressure builds up higher than between 1900 and 2200 psig.

4.6.3 Hydraulics

4.6.3.1 Hydraulic Pump I Z 1

Hydraulic power is generated by a variable displacement hydraulic pump. The APU drives the pump at 3800 rpm ± 8 percent nominal speed to produce a full-flow pump output of 3000 to 3250 psig at a flow of 55 gpm. The speed is controlled to an accuracy of ± 8 percent of the selected speed due to the inertia of the turbine wheel. Hydraulic fluid from the hydraulic reservoir is supplied to the hydraulic pump inlet at 55 to 75 psig.

Each pump will provide hydraulic power to move the SRB TVC actuators. HPU A and HPU B supply redundant hydraulic power to each of the actuators.

4.6.3.2 Reservoir F Z 1

Each HPU has a hydraulic fluid reservoir. The reservoir provides pressurized fluid to the hydraulic pump inlet by means of a differential area piston and a mechanical spring which compensate for fluid volumetric changes resulting from temperature and/or system operating conditions.

The reservoir has a high pressure chamber, a low pressure chamber, and an air chamber. The fluid from the high pressure chamber has a pressure of 3000 to 3250 psig and is routed to the servoactuators. Low pressure (55 to 75 psig) fluid is returned from the servoactuators to the low pressure chamber. During flight operations, the high pressure chamber uses flow tapped off the output of the pump to actuate the differential area piston which maintains the low pressure chamber operating pressure required for the pump. The mechanical spring regulates the response to system pressure changes. The air chamber is enclosed and vented to the atmosphere to eliminate a positive or negative back pressure reaction against the piston. The vent port fitting has a 100μ filter to protect the piston seal from contamination. Bleed valves are located on the high point of the low pressure and high pressure chambers to allow removal of trapped air prior to flight.

A low pressure relief valve is provided to protect against GSE overfill/ overpressurization. Overpressurization (3650 psig or greater) in the high pressure chamber is relieved by a relief valve which will open and permit fluid to flow to the low pressure chamber.

The fluid quantity indicator is a hermetically sealed unit which senses fluid quantity as a function of the reservoir piston position. The indicator provides an electrical output signal that is directly proportional to the fluid level in the reservoir.

4.6.3.3 Accumulator G Z 3

The hydraulic pressure accumulator dampens subsystem pressure impulses and minimizes pump supply pressures ripples. The accumulator has a volume of 49 in^3 for hydraulic fluid, and 98 in^3 for GN_2.

4.6.3.4 Hydraulic Fluid Check Valve and Filter H Z 2

A check valve and filter are located in the hydraulic line between the hydraulic pump and the servoactuators. The check valve blocks backflow into the hydraulic pump during system bleed and fill, and the filter removes nonsoluable pollutants that are larger than 5 microns from the hydraulic fluid.

References

Solid Rocket Booster Thrust Vector Control Subsystem Description, Nasa TM-82546, September 1983.

Space Shuttle Systems Handbook, NASA-JSC-11174, November 30, 1987.

Space Shuttle Probabilistic Risk Assessment - APU/HPU VOLUME III, MDAC WORKING PAPER 1.0-WP-VA88004-03, December 18, 1987.

TABLE 4.6-I.- BASIC AREAS OF OPERATION DIFFERENCE

	Orbiter	Booster
Nominal power	135 HP	135 HP
Mission length	75 minutes - EI-13 start 95 minutes - TIG-5 start	166 seconds
Mission fuel consumption	175 lb. - EI-13 start 200 lb. - TIG-5 start	6.6 lb - normal mission 10.0 lb - back-up mission
Operational environment	-65° F space ambient	-salt water exposure -increased vibration loads -water entry at pressure loads -limited soakback time
Handling differences	Requires refurbishment after 20 hours Oil and oil filter changed each flight	-requires refurbishment between missions -simplified controls -reduced instrumentation -N_2 checkout capability
Thermal	-oil and water coolers -thermally insulated	-no oil cooling -no water coolers -uninsulated -eliminated fuel pump heater
Fuel isolation valve	Two parallel valves	One valve
APU speed control		
normal	-103 percent	-100 percent
backup	-113 percent	-110 percent
second backup	-115 percent	-112 percent
overspeed protection	Yes (will AUTO-SHUTDOWN)	No (containment ring only)
underspeed shutdown	Yes (will AUTO-SHUTDOWN)	No

5.1 MASTER EVENTS CONTROLLER OVERVIEW

5.1.1 General

The master events controller (MEC) is the interface unit between the general purpose computers (GPC's), and the orbiter, ET, and SRB pyrotechnic and control devices. The function of the MEC is to receive digital commands from the GPC's and to buffer, compare, and execute those commands through sequential hardware during launch, SRB separation, and ET separation. The MEC subsystem operating program (SOP) converts sequential event commands from the redundant set to a digital command format. Each event command is addressed, classified according to criticality (either critical (CR) or noncritical (NCR)), and output simultaneously by the four computers. The GPC's transmit commands to the MEC's via four of the flight-critical data buses (FC 5, 6, 7, and 8).

Each MEC is redundantly powered by all three main dc buses (28 V dc). The MEC-1 and MEC-2 power switches on panel 017 provide manual power control using remote power controllers (RPC's).

The MEC uses the built-in test equipment (BITE) to monitor the status of the pyro initiator controllers (PIC's). Prelaunch testing includes PIC load tests using a dummy NASA standard initiator (NSI) and PIC resistance tests using an operational NSI for system interface checkout. This prelaunch checkout evaluates MEC internal functions and performance, including internal power supplies, parity errors, and command processing.

All the MEC functions are carried out in launch (Ops 1), SRB separation (Ops 1), and ET separation (Ops 1 or 6) sequences. During launch, in a precisely timed sequence, the MEC arms and fires the PIC's for the following functions: SRM ignition, SRB holddown release, T-0 umbilical retraction, and the ET GH_2 vent arm retraction.

The MEC also controls the process of SRB staging. After the chamber pressure drops below 50 psi in both SRB's, the SRB range safety system is powered down and safed, and the recovery system is activated. At the appropriate time, the SRB TVC's are deadfaced, the bolts at the attachment points are severed, and the SRB separation motors are ignited.

At MECO, the MEC controls the ET separation sequence. The tumble valve is opened by a single pyro function (MEC 2 only), the ET umbilicals are unlatched and retracted, and the bolts at the attachment points are severed. After ET separation, the MEC functions are complete for the mission and the MEC's are powered down.

5.2 MASTER EVENTS CONTROLLER

5.2.1 General

The MEC is a line replaceable unit (LRU) that provides for the transfer and signal conditioning of control and measurement data between the GPC's and the orbiter, ET, and SRB pyrotechnic and control devices. This section provides a hardware description of the MEC based on its functional requirements.

A description of various MEC interfaces (i.e., data bus, power, cooling, and pyrotechnical) is provided in section 5.2.2. A description of the MEC internal configuration is provided in section 5.2.3. The MEC currently flown is a type IV MEC with added redundancy (i.e., FIRE 3 commands).

For detailed information schematics, refer to the following Space Shuttle Systems Handbook (SSSH) drawings:

- 10.1-1 - MEC Command Table
- 10.1-2 - MEC Overview
- 10.1-3 - Master Events Controller
- 10.1-4 - Master Events Controller

5.2.2 MEC Interfaces

The various MEC hardware and digital interfaces are described in the following sections.

5.2.2.1 MEC Input Power Interface

Each MEC is redundantly powered by all three main dc buses (28 V dc). The MEC-1 and MEC-2 power switches on panel 017 provide manual power control using remote power controllers (RPC). The MEC power inputs schematic is shown in figure 5.2-1.

Within each MEC, power from power supplies 1 and 2 is routed through the redundant paths of Core A and Core B, respectively. The maximum power demand of a MEC, per the procurement spec, is 260 watts at 28 V dc with 50 percent of all drivers in the logic "one" or ON state and the four multiplex interface adaptors (MIA's) in the receive mode.

5.2.2.2 Cooling System Interface

A coldplate is installed between the MEC and the shelf on which the unit is mounted to provide temperature control. The coldplate covers the entire mounting surface of the MEC.

5.2.2.3 Data Bus Interface

Each MEC contains four MIA's to receive digital command words and command data words from a corresponding primary avionics software system (PASS) GPC via one of the flight critical data buses (FC 5 or 6 for MEC 1, FC 7 or 8 for MEC 2). If the backup flight system (BFS) is engaged, all four data buses are controlled by the single BFS GPC. the MEC logic circuits interface with each MIA to receive input commands and to transmit return data via the data buses. Reference SSSH dwgs 10.1 and 10.2.

The MEC hardware functions are controlled by the MEC SOP. The principle function of the MEC SOP is to process user function discrete commands to generate the following serial-digital signals to the MEC: MEC CR commands, MEC NCR commands, and MEC master reset commands. The MEC SOP is processed on demand as addressed, classified according to criticality, and output simultaneously by four computers. Refer to the Booster Systems Software Handbook 2.8 for a description of the MEC SOP.

A given MEC responds only to commands with its own address. Two types of commands control MEC functions: critical commands and noncritical commands. Each time the GPC sends a command, it has certain basic information in the command word, and the additional command data words that follow are processed based on the instructions in the command word. When a CR command is announced in the command word, only a CR command will be processed from a succeeding command data word. Conversely, if a NCR command is announced a CR command will be rejected in that data block of the computer. The command message structure schematic is shown in figure 5.2-2.

CR and NCR commands require comparison of data on the four channels corresponding to the flight critical data buses. At least two coded critical words must compare identically for an output command from the MEC. If two of the four command words compare, and the remaining two coded words compare, but the two pairs miscompare, this miscompare shall inhibit processing and set the BITE register. NCR commands are transmitted on channels 1, 2, and 3 in core A and channels 1, 3, and 4 in core B. The MEC channel Boolean voter logic for NCR circuits is 2AND(1OR3) for core A and 3AND(1OR4) for core B. The voting logic for CR and NCR commands are shown in figure 5.2-5."

The information flow on the data buses is comprised of three types of 28-bit words: command word, command data word, and response data word. A command word from the GPC to the MEC is comprised of a 3-bit command synchronization waveform field, a 5-bit MEC address, a 19-bit control field, and a parity bit. A command data word from the GPC to the MEC is comprised of a 3-bit data synchronization waveform field, a 5-bit MEC address, a 16-bit data field, a 3-bit check pattern, and a parity bit. A response data word from the MEC to the GPC is comprised of a 3-bit data sychronization waveform field, a 5-bit MEC address, a 16-bit data field, a power transient flag bit, an error flag bit, a validity flag bit, and a parity bit. The response data word from the MEC closes the loop for the computer, and, if the response is not present, a second request is issued. Data requested by a GPC on one data bus has the return response transmitted by the MEC on the same data bus. (Note: All response data are transmitted preflight. No data flows from the MEC to the GPC during flight.) The command and data work bit structure schematic is shown in figure 5.2-2.

The MEC CR command inputs and outputs are shown in table 5.2-I. The four digit hexadecimal code word represents a unique 16-bit command data field to the HEX decoder. All 16 binary digits must be correct or no plausible output function is produced. Although a command's HEX and binary code is the same in MEC 1 and MEC 2, the command meaning will differ between the two MEC's.

If two or more functions are time coincident, a single MEC input signal can produce an output to several devices; e.g., LT SRB IGNITION ARM, RT SRB IGNITION ARM, AND SRB HOLDDOWN ARM are produced from the input command SRM IGN ARM from the R/S LCH SEQ SOP.

The NCR command input signals to the MEC are shown in table 5.2-II. The NCR commands are represented by a single bit or discrete. The 16-bit data field of the NCR command data words are, therefore, capable of changing 16 commands at one time. The NCR commands are stored in a command memory that provides continuous outputs for control functions. Redundancy of systems controlled by NCR commands is maintained by having MEC 1 controlling power to one system and MEC 2 controlling power to the second system; e.g., MEC 1 turns on power A and MEC 2 turns on power B to the left SRB.

In the original MEC design, pyro functions were executed by the CR commands and nonpyro functions were executed by the NCR commands. The updated MEC described in this document required certain readjustments to the original design to accommodate the addition of FIRE 3 commands for improved redundancy. The FIRE 3 commands (SRB IGNITION, T-O UMBILICAL SEP, and ET/ORB SEP) were added to the NCR command register, and the ET TUMBLE ARM AND FIRE, SRB RSS SAFE, and SRB ATVC (electrical) deadface commands were moved to the CR command register. Four signals are use to complete a pyro firing event: ARM, FIRE 1, FIRE 2, and FIRE 3, with the FIRE 3 (NCR) command enabling the FIRE 2 (CR) command.

5.2.2.4 Pyrotechnic Interface

Eight PIC's are assembled in the MEC. The MEC drives eight additional PIC's located in the orbiter and SRB's. The eight PIC's in the MEC drive the ET/orbiter sep and umbilical unlatch pyro devices. The eight additional PIC's drive the left and right SRB ignition and separation pyro devices. The PIC's for the T-0 holddowns and umbilicals are all part of the MLP. The ET tumble valve has no PIC. The PIC'S are used to evaluate three separate commands: ARM, FIRE 1, and FIRE 2s (FIRE 2 is enabled by FIRE 3 commands in core A and core B logic). When all are valid or true, it will fire an NSI. The PIC's have a standard design that is used throughout the orbiter. Each PIC module is unpowered until the ARM command is sent. The ARM command is a 20 to 32 V dc signal which is actually input power for the PIC's dc/dc converter (fig. 5.2-3), which in turn will charge a capacitor bank to store sufficient energy to fire an NSI.

A 37.5 V dc charge is required on the PIC capacitors to satisfy the computer voltage check. An analog of the capacitor charge is converted to digital form by the MEC's and becomes a data response to the GPC. The FIRE 1 signal is filtered and used to enable the NSI return path to the capacitor bank. The FIRE 2 signal must also be present, however, to enable this return path. The FIRE 2 signal has an input noise filter, and, when it is present, two functions relating to the capacitor discharge takes place: an inhibit is generated for the charging path of the capacitors and the capacitor bank is connected to the NSI allowing the NSI to perform its function.

Verification of the PIC's and their corresponding NSI's is accomplished by the PIC's. The PIC's incorporate circuitry for a pyro-resistance test and a pyro-load test. The resistance test is a continuity check of the output circuit, including the NSI. The load test is an evaluation of the PIC output signal with a dummy NSI in place. Both are preflight checks only. An analog output of the capacitor bank voltage is supplied to the MEC and monitored by the RSLS. After the ignition PIC ARM commands are issued at T-18 seconds, the RSLS will initiate a launch "hold" if any SRB ignition PIC voltage is less than 35.7 V and greater than 40.5 V.

Each pyro function uses two NSI's for redundancy, with one NSI driven by MEC-1 and the other NSI by MEC-2. (An exception is the ET tumble valve, which is opened by one NSI and pyro driven by MEC-2.) A list of pyro functions is shown in figure 5.2-4.

5.2.2.5 Nonpyro Output Interface

The following nonpyro functions are also performed by the MEC's:

a. SRB bus power
b. SRB RSS power
c. SRB RSS safe
d. SRB TVC deadfacing

These functions are operated by power relays, which in turn are driven by redundant 28 V dc signals from MEC 1 and MEC 2. An exception is the SRB ATVC 2 and 4 and 1 and 3 which are driven by MEC's 1 and 2, respectively. Reference figure 5.2-4 for the functional diagram.

5.2.3 MEC Internal Configuration

An MEC functional block diagram is provided in figure 5.2-5. Each of the two MEC's are functionally divided into two cores of the same internal design. Each core contains four identical input channels for data processing that interface with the four MIA's. Outputs from the two cores (A and B) with the same function are OR'ed together to provide a single output signal. Cores A and B are generally related to channels 1/2 and 3/4, respectively, and each core has its own power source.

The MEC is designed with CR and NCR circuits separated functionally on individual modules. The power source for CR and NCR circuits of a core originates from separate power converter outputs. All logic clocking sources for CR and NCR circuits of a core originate from separate MIA's.

The command words throughput by the MIA's are decoded and tested for format and address in the load controllers, where command type (CR or NCR) of subsequent command data is determined. CR command data (in the form of hexadecimal coded command data words) are first stored in a skew holding register for 4.45 microseconds. After the skew holding period, the data are compared to the data in the other three channel registers by the critical command 2/4 voter. A minimum two-out-of-four bite is required to execute a GPC critical command. Critical command functions can only be reset by a master responsible for proper clocking of the critical command voter and decoder functions. The selected CR commands are stored in a register prior to decoding, with the decoded command subsequently setting the appropriate registers.

The NCR signals are used to set and/or reset NCR output drivers and selected, combined CR and NCR output drivers. NCR command data words are stored in the NCR command registers and then compared in the Cores A and B data voters. A minimum two-out-of-three vote is required to execute a GPC NCR command. NCR commands are not affected by the master reset but are "zero-set" at power up. The NCR data registers are disabled 4.42 milliseconds following an NCR output driver command. Cores A and B NCR voters control reset and clocking for the NCR commands. Each bit in the data field of the NCR command data words controls a separate NCR driver and function. The selected NCR commands are stored in a register prior to setting the appropriate driver.

Protection against inadvertent and/or premature generation of FIRE 2 critical output signals has been provided by separating circuits and adding the FIRE 3 redundancy. Figures 5.2-7 and 5.2-8 illustrate the logic for the majority of the PIC logic circuits wherein the ARM and FIRE 1 signals from cores A and B are OR'ed together and routed to the MEC internal or external PIC's. The FIRE 2 signal in this logic scheme is the product of AND'ing the critical MEC FIRE 2 signals and the NCR MEC FIRE 3 signals within each core to prevent the premature initiation of the FIRE 2 signal.

Figure 5.2-9 illustrates the special logic for the ET/orbiter separation and ET umbilical unlatch circuits. The ARM and FIRE 1 signals from cores A and B are OR'ed together and routed to the internal and external PIC's. The FIRE 2 PIC signal in this logic is the product of the AND'ing of the FIRE 1, 2, and 3 signals from each core and OR'ing the results prevents a premature FIRE 2 signal.

Figure 5.2-10 illustrates the logic for the ET umbilical retract voting and interlock circuit. Only one output signal is required from each core. Each signal is generated as the AND'ed product of each core's FIRE 1, FIRE 2, and FIRE 3 signals. The core A and core B signals are OR'ed together as a single umbilical retract signal at the MEC output. The individual AND'ing of each core's signal prevents premature initiation of the ET umbilical retract function.

Prior to STS 41-D, an SRB SEP/MEC SOP timing problem associated with the redesigned MEC was discovered that could have resulted in a 1-in-24 chance of SRB SEP failing to occur. This problem was the combined result of the possible failure of the SRB SEP FIRE 3 command to be output the first time it is set (because of an output priority problem) and the removal of a bleed resistor in the isolation driver circuits (IDC) immediately upstream of the transient protection circuits (TPC) where the FIRE 2 commands are enabled by their respective FIRE 3 commands (fig. 5.2-11). This problem was resolved on STS 41-D by revising the PASS software to raise the output priority of the FIRE 3 commands and reinstalling the bleed resistor.

The MEC contains BITE for detecting, recording, and verifying MEC failures or out-of-tolerance conditions that affect MEC operations. When an internal discrepancy occurs, the MEC generates an error signal that sets a bit flag in a BITE register. When the GPC requests MEC BITE status, the BITE register contents are transmitted to the GPC and the BITE register is reset to logic zero for a new status record.

The MEC BITE preflight testing includes the PIC load tests using a dummy NSI, and PIC resistance tests using an operational NSI for system interfacer checkout. In-flight MEC BITE testing includes parity, bit count, and transmission errors at the MIS's, voltage verifications (5 V dc and 12 V dc) at the power supplies, and PIC operational verification.

The MEC module configuration is shown in figure 5.2-7. There are seven module types, with one or more of each module type used in the MEC.

TABLE 5.2-I.- MEC SOP CRITICAL COMMAND INPUTS AND OUTPUTS

MEC SOP input name	MSID	Source	MEC SOP OUTPUT HEX Code	Name	MSID	MEC H/W output destination	Characteristics
T-O UMB ARM	V90X8407X	R/S LCH SEQ	CEDC CEDC	MEC1 L T-O UMB REL ARM MEC2 R T-O UMB REL ARM	V76K4609B V76K4613B	T-O UMB SEP GSE	28V,20mA
T-O UMB FIRE 1	V90X8408X	R/S LCH SEQ	CE6A CE6A	MEC1 L T-O UMB REL FIRE 1 MEC2 R T-O UMB REL FIRE 1	V76K4610B V76K4614B	T-O UMB SEP GSE	28V 20mA
T-O UMB FIRE 2/3	V90X8698X	R/S LCH SEQ	CE9A CE9A	MEC1 L T-O UMB REL FIRE 2 MEC2 R T-O UMB REL FIRE 2	V76K4611B V76K4615B	T-O UMB SEP GSE	28V,20mA
SRM IGN ARM	V90X8404X	R/S LCH SEQ	3EAC 3EAC	MEC1 SRM IGN ARM MEC2 SRM IGN ARM	V76K6951B V76K6961B	L SRB IGN R SRB IGN HOLD DOWN	28V,10mA 28V,10mA 28V,10mA
SRM IGN FIRE 1	V90X8405X	R/S LCH SEQ	3E6A 3E6A	MEC1 SRM IGN FIRE 1 MEC2 SRM IGN FIRE 2	V76K6953B V76K6963B	L SRB IGN R SRB IGN HOLD DOWN	28V,10mA 28V,10mA 28V,10mA
SRM IGN FIRE 2/3	V90X8699X	R/S LCH SEQ	3E42 3E42	MEC1 SRM IGN ARM MEC2 SRM IGN ARM	V76K6954B V76K6964B	L SRB IGN R SRB IGN HOLD DOWN	28V,10mA 28V,10mA 28V,10mA
SRM SEP ARM	V90X8335X	SRB SEP SEQ	3154 3154	MEC1 SRB SEP ARM MEC2 SRB SEP ARM	V76K6956B V76K6966B	L SRB SEP R SRB SEP	28V,10mA 28V,10mA
SRM SEP FIRE 1	V90X8341X	SRB SEP SEQ	316A 316A	MEC1 SRB SEP FIRE 1 MEC2 SRB SEP FIRE 1 & RCVY ON	V76K6958B V76K6968B	L SRB SEP R SRB SEP	28V,10mA 28V,10mA
SRM SEP FIRE 2/3	V90X8354X	SRB SEP SEQ	319B 319B	MEC1 SRB SEP FIRE 2 MEC2 SRB SEP FIRE 2 & RCVY ARM	V76K6959B V76K6969B	L SRB SEP R SRB SEP	28V,10mA 28V,10mA
SRM RSS SAFE	V90X8337X	SRB SEP SEQ	EB68 EB95 EB68 EB95	MEC1 RSS L SAFE 1 MEC2 RSS R SAFE 2 MEC2 RSS R SAFE 1 MEC2 RSS L SAFE 2	V76K7508B V76K7509B V76K7608B V76K7609B	SRB RSS SAFE 1 SRB RSS SAFE 2 SRB RSS SAFE 1 SRB RSS SAFE 2	28V,10mA 28V,10mA 28V,10mA 28V,10mA

TABLE 5.2-I.- Continued

MEC SOP input name	MSID	Source	MEC SOP OUTPUT HEX Code	Name	MSID	MEC H/W output destination	Characteristics
ATVC SRB 26V DEAD FACE	V90X8339XA V90X8339XB	SRB SEP SEQ ET SEP SEQ	EC68 EC68 EC90 EC90	MEC1 ATVC 2 SRB 26V DEAD FACE MEC2 ATVC 1 SRB 26V DEAD FACE MEC1 ATVC 4 SRB 26V DEAD FACE MEC2 ATVC 3 SRB 26V DEAD FACE	V76K7013B V76K7113B V76K7014B V76K7114B	ATVC 2 26V DEAD FACE ATVC 2 26V DEAD FACE ATVC 4 26V DEAD FACE ATVC 3 26V DEAD FACE	28V,10mA 28V,10mA 28V,10mA 28V,10mA
ET UMB UNLCH ARM	V90X8247X	ET SEP SEQ	C121 C121	MEC1 ET UMB UNLCH ARM MEC2 ET UMB UNLCH ARM	V76K4617B V76K4618B	L UMB UNLCH PICS 1,2,3 R UMB UNLCH PICS 4,5,6	28V,1.5A 28V,1.5A
ET UMB UNLCH FIRE 1	V90X8256X	ET SEP SEQ	C162 C162	MEC1 ET UMB UNLCH FIRE 1 MEC2 ET UMB UNLCH FIRE 1	V76K4619B V76K4620B	L UMB UNLCH PICS 1,2,3 R UMB UNLCH PICS 4,5,6	28V,30mA 28V,30mA
ET UMB UNLCH FIRE 2/3	V90X8242X	ET SEP SEQ	C193 C193	MEC1 ET UMB UNLCH FIRE 2 MEC1 ET UMB UNLCH FIRE 2	V76K4623B V76K4624B	L UMB UNLCH PICS 1,2,3 R UMB UNLCH PICS 4,5,6	28V,30mA 28V,30mA
ET UMB RETR FIRE 1	V90X8263X	ET SEP SEQ	7463 7463	MEC 1 ET UMB RETR FIRE 1 MEC 1 ET UMB RETR FIRE 1	V76K4655B V76K4659B	L UMB SOL BUFRS 1,2,3 R UMB SOL BUFRS 1,2,3	ENABL FIRE2 ENABL FIRE2
ET UMB RETR FIRE 2/3	V90X8243X	ET SEP SEQ	7498 7498	MEC1 ET UMB RETR FIRE 2 MEC2 ET UMB RETR FIRE 2	V76K4656B V76K4660B	L UMB SOL SW. 1,2,3 R UMB SOL SW. 1,2,3	28V,1.5A 28V,1.5A
ET/ORB SEP ARM	V90X9265X	ET SEP SEQ	E117 E117	MEC1 ET/ORB SEP ARM MEC2 ET/ORB SEP ARM	V76K6909B V76K6911B	STRUCT SEP PICS 7, 8 AND FWD PIC	28V,1.5A
ET/ORB SEP FIRE 1	V90X8244X	ET SEP SEQ	E168 E168	MEC1 SRB SEP FIRE 1 MEC2 SRB SEP FIRE 1	V76K6913B V76K6915B	STRUCT SEP PICS 7, 8 AND FWD PIC	28V,1.5A
ET/ORB SEP FIRE 2/3	V90X8241X	ET SEP SEQ	E199 316A	MEC1 SRB SEP FIRE 2 MEC2 SRB SEP FIRE 2	V76K6914B V76K6916B	STRUCT SEP PICS 7, 8 AND FWD PIC	28V,1.5A
ET TUMBLE ARM	V90X8251X	ET SEP SEQ	EBC2 EBC2	MEC1 ET TUMBLE ARM MEC2 ET TUMBLE ARM	V78K7504B V76K7604B	ET tumble arm	28V,10mA
ET TUMBLE FIRE	V90X8252X	ET SEP SEQ	ECB6 ECB6	MEC1 ET TUMBLE ARM MEC2 ET TUMBLE ARM	V78K7505B V76K7605B	ET tumble arm	28V,10mA

TABLE 5.2-II.- MEC SOP NON-CRITICAL COMMANDS INPUT AND OUTPUTS

MEC SOP input name	MSID	Source	MEC SOP OUTPUT Name	CMD	MSID	MEC output destination	Characteristics
MEC 1 & 2 MASTER RESET	V90X8258XC	R/S LCH SEQ	(A) MEC1 MASTER RESET		V76K7098B		
	V90X8258XB	SRB SEP SEQ	(A) MEC2 MASTER RESET		V76K7198B		
	V90X825BXA	ET SEP SEQ					
T-O UMB FIRE 2/3	V90X8698X	R/S LCH SEQ	13 MEC1 L T-O UMB REL FIRE 3	D	V76K4612B	ENABLES FIRE 2	28V,20mA
			13 MEC2 R T-O UMB REL FIRE 3	D	V76K4616B		
SRM IGN FIRE 2/3	V90X8699X	R/S LCH SEQ	12 MEC1 SRM IGN FIRE 3	C	V76K6955B	ENABLES FIRE 2	28V,20mA
			12 MEC2 SRM IGN FIRE 3	C	V76K6965B		
SRB SEP FIRE 2/3	V90X8354X	SRB SEP SEQ	16 MEC1 SRB SEP FIRE 3	G	V76K6960B	ENABLES FIRE 2	28V,20mA
			16 MEC2 SRB SEP FIRE 3	G	V76K6970B		
SRB RSS POWER OFF	V90X8336X	SRB SEP SEQ	14 MEC1 RSS L SRB PWR OFF	E	V76K7006B		28V,10mA
			15 MEC1 RSS R SRB PWR OFF	F	V76K7007B		28V,10mA
			14 MEC2 RSS L SRB PWR OFF	E	V76K7106B		28V,10mA
			15 MEC2 RSS R SRB PWR OFF	F	V76K7107B		28V,10mA
ATVC SRB IVD POWER ON	V90X8338XA	SRB SEP SEQ	21 MEC1 ATVC 2 SRB IVD A PWR ON	L	V76K7013B		28V,10mA
	V90X8338XB	ET SEP SEQ	22 MEC1 ATVC 4 SRB IVD B PWR ON	M	V76K7014B		28V,10mA
			21 MEC2 ATVC 1 SRB IVD C PWR ON	L	V76K7113B		
			22 MEC2 ATVC 3 SRB IVD B PWR ON	M	V76K7114B		
SRB PWR ON	V90X8343X	ET SEP SEQ	10 MEC1 L SRB PWR A	A	V76K7002B		28V,10mA
			11 MEC1 R SRB PWR A	B	V76K7003B		28V,20mA
			19 MEC1 L SRB PWR C	J	V76K7011B		28V,10mA
			20 MEC1 R SRB PWR C	K	V76K7012B		28V,10mA
			10 MEC2 L SRB PWR B	A	V76K7102B		28V,10mA
			11 MEC2 R SRB PWR B	B	V76K7103B		28B,20mA
			19 MEC2 L SRB PWR C	J	V76K7111B		28V,10mA
			20 MEC2 R SRB PWR C	K	V76K7112B		28V,10mA
ET DFI PWR ON	V90X8255X	ET SEP SEQ	18 NO OUTPUT	I	N/A	N/A	
ET UMB UNLCH FIRE 2/3	V90X8242X	ET SEP SEQ	17 MEC1 ET UMB UNLCH FIRE 3	H	V76K4625B	ENABLES FIRE 2	28V,30mA
			17 MEC2 ET UMB UNLCH FIRE 3	H	V76K4626B		
ETUMB RET FIRE 2/3	V90X8243X	ET SEP SEQ	23 MEC1 ET UMB RETR FIRE 3	N	V76K4657B	ENABLES FIRE 2	28V,30mA
			23 MEC2 ET UMB RETR FIRE 3	N	V76K4661B		
E/ORB SEP FIRE 2/3	V90X8241X	ET SEP SEQ	24 MEC1 ET UMB RETR FIRE 3	O	V76K6921B	ENABLES FIRE 2	28V,30mA
			24 MEC2 ET UMB RETR FIRE 3	O	V76K6922B		

(A) Resets the following bits: 12, 13, 16, 17, 12, 24

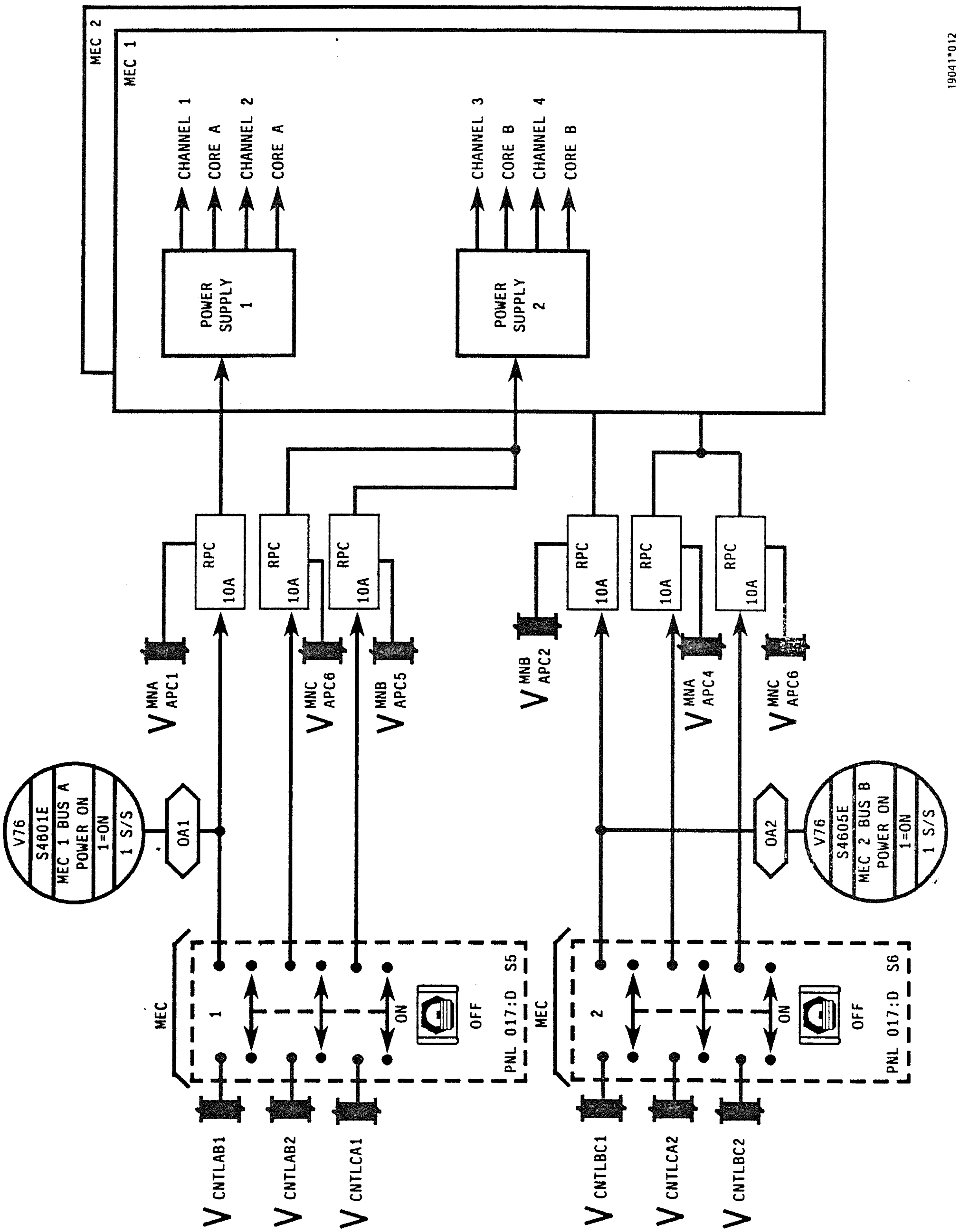
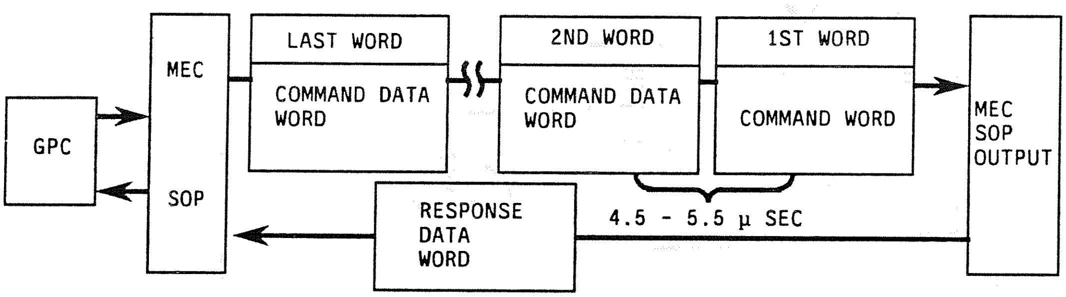


Figure 5.2-1.- Power output schematic.

MEC COMMAND MESSAGE STRUCTURE



COMMAND WORD FORMAT

	CMD SYNC FIELD	MEC ADDRESS FIELD	MODE CONTROL FIELD	MODULE ADDRESS FIELD	CHANNEL ADDRESS FIELD	NO. OF WORDS FIELD	PARITY
BIT NO.	1 – 3	4 – 8	9 – 13	14 – 17	18 – 22	23 – 27	28

COMMAND DATA WORD FORMAT FROM GPC TO MEC

	DATA SYNC FIELD	MEC ADDRESS FIELD	DATA FIELD (16 BITS)	1	0	1	PARITY
BIT NO.	1 – 3	4 – 8	9 – 24	25	26	27	28

RESPONSE DATA WORD FORMAT FROM MEC TO GPC

	DATA SYNC FIELD	MEC ADDRESS FIELD	DATA FIELD (16 BITS)	S 1	E 0	V 1	PARITY
BIT NO.	1 – 3	4 – 8	9 – 24	25	26	27	28

19041*007

Figure 5.2-2 - Command message format.

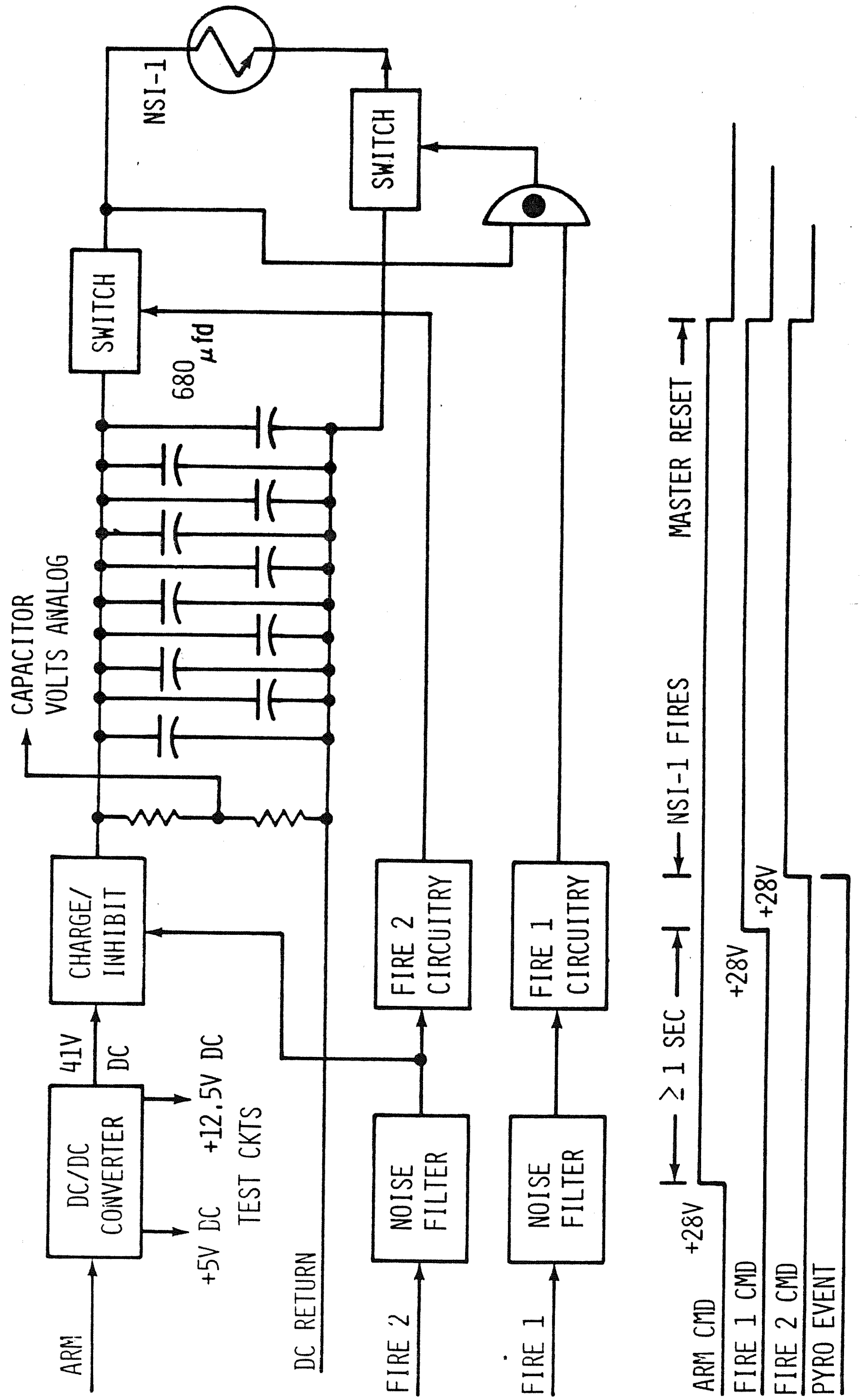
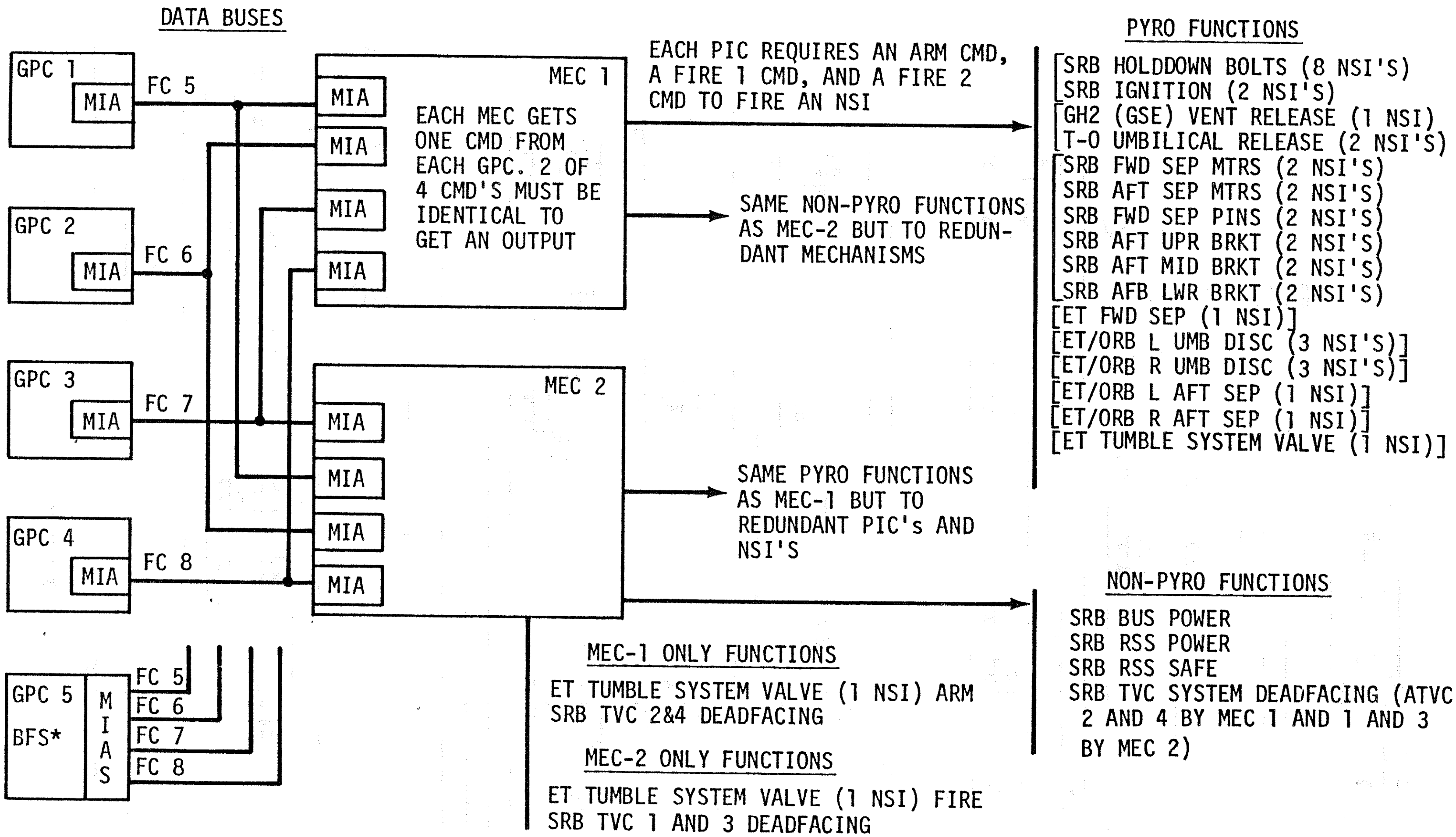


Figure 5.2-3.- Pyrotechnic initiator controller.



*IF THE BFS IS ENGAGED, THAT ONE GPC COMMANDS OVER ALL FOUR DATA BUSES. THE MEC'S CANNOT TELL WHETHER THE PASS OF THE BFS IS SENDING THE CMDS.

PIC = PYRO INITIATOR CONTROLLER
NSI = NASA STANDARD INITIATOR
[ ] = IMPLIES A SINGLE CMD OUT OF THE MEC GOES TO ALL THESE DEVICES.

Figure 5.2-4.- Master events control functional diagram.

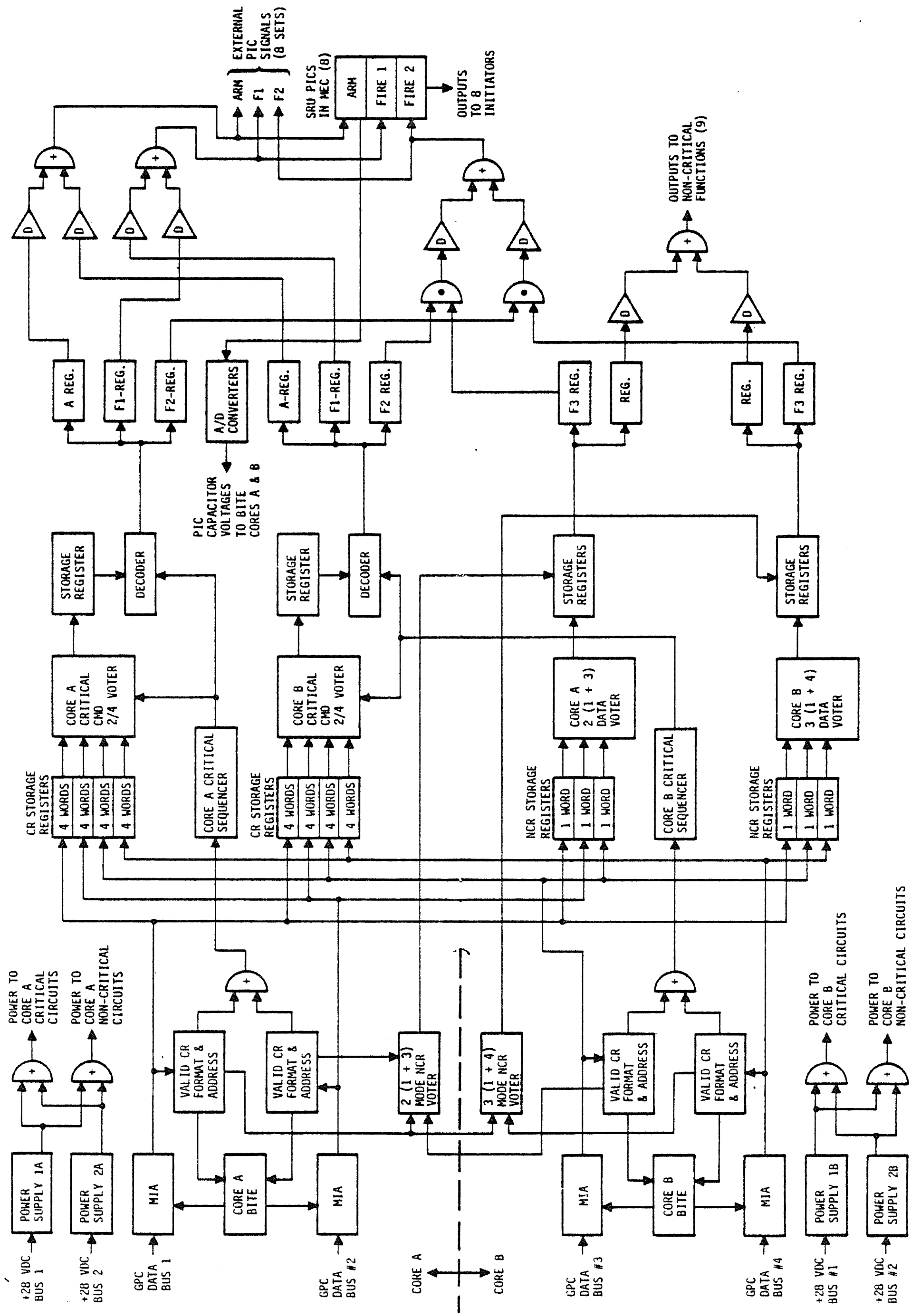
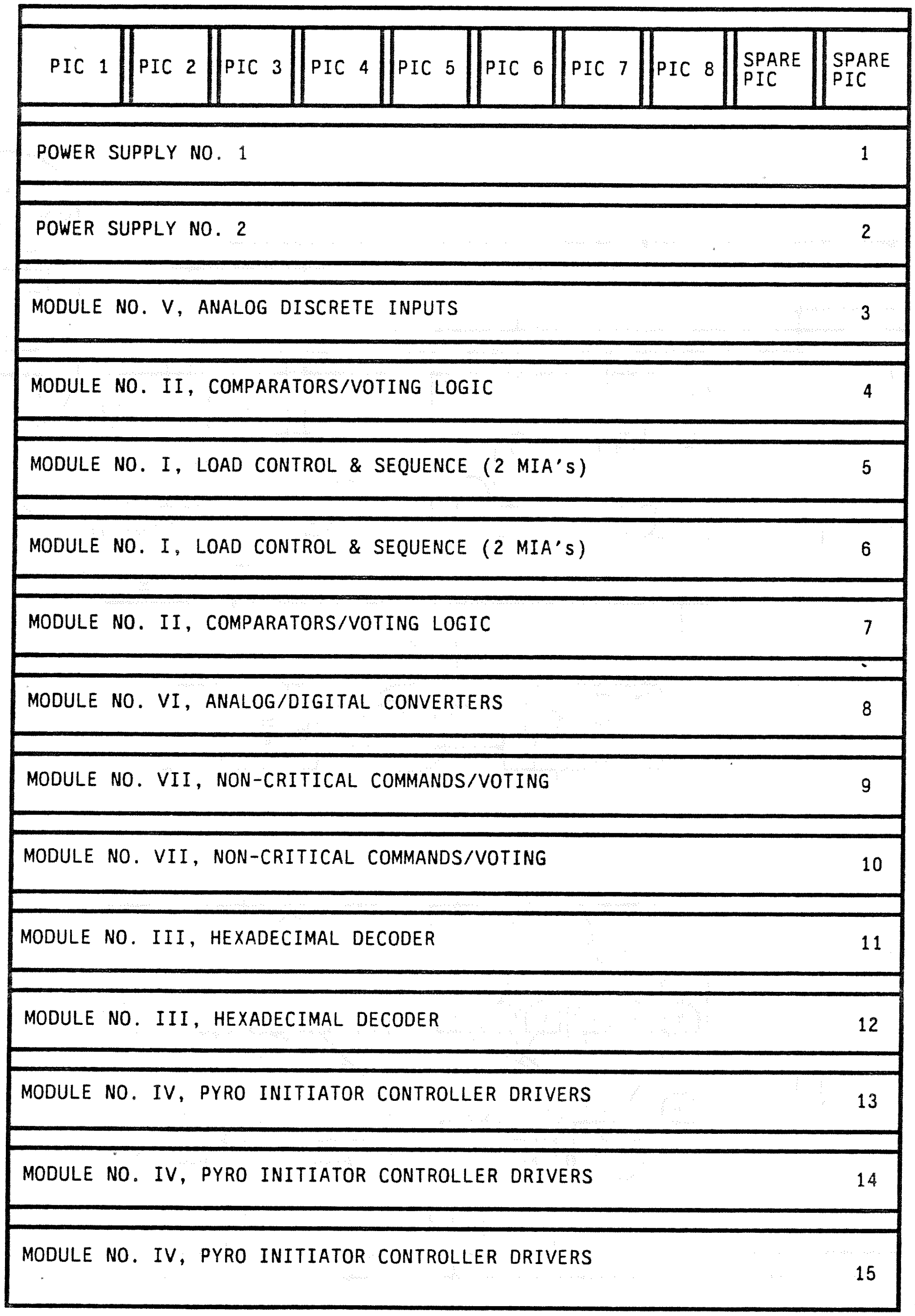


Figure 5.2-5.- MEC functional block diagram.



19041*008

Figure 5.2-6 - MEC module configuration.

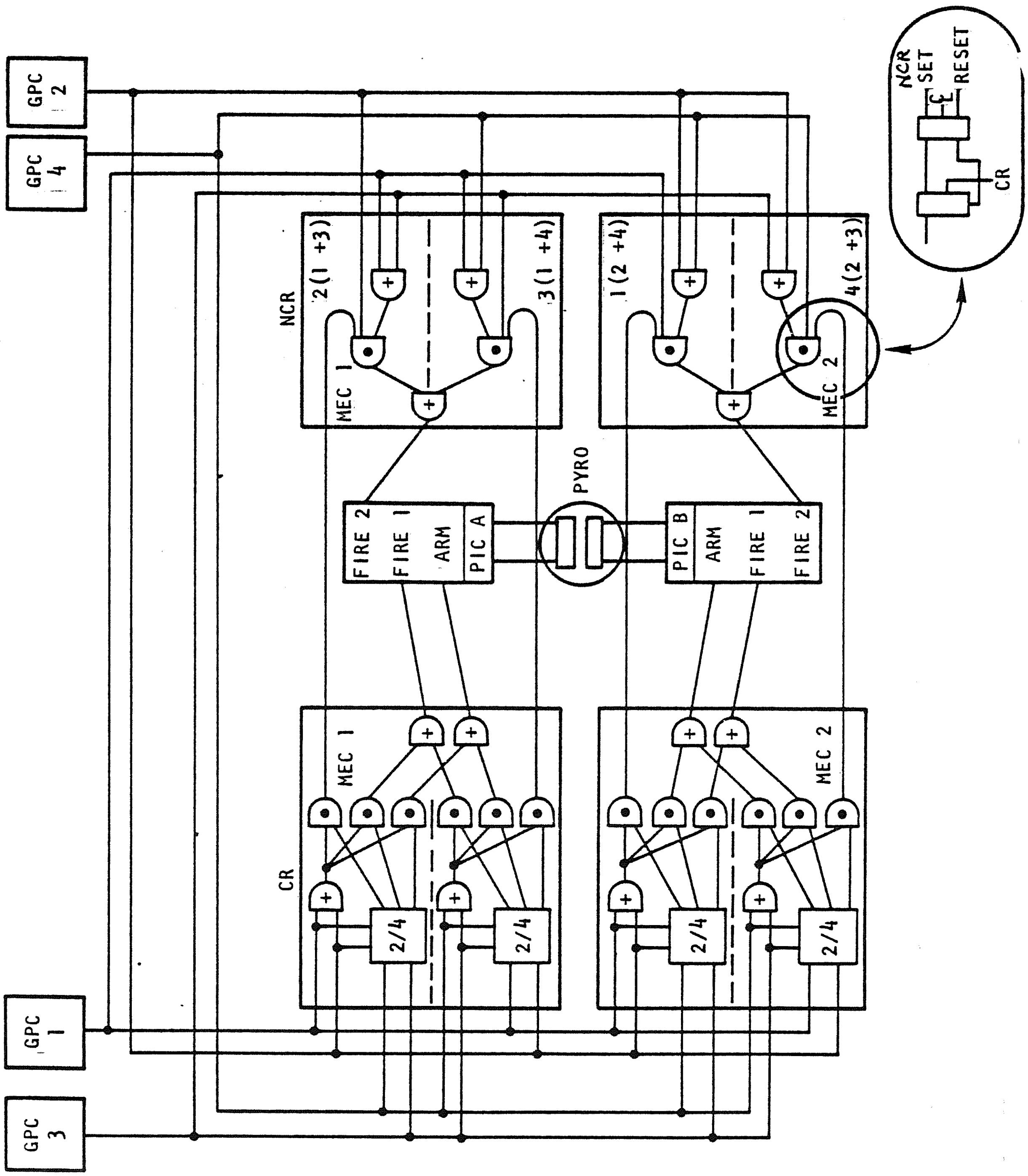


Figure 5.2-7.- Critical commands signal paths, including non-critical Fire 2 qualifications.

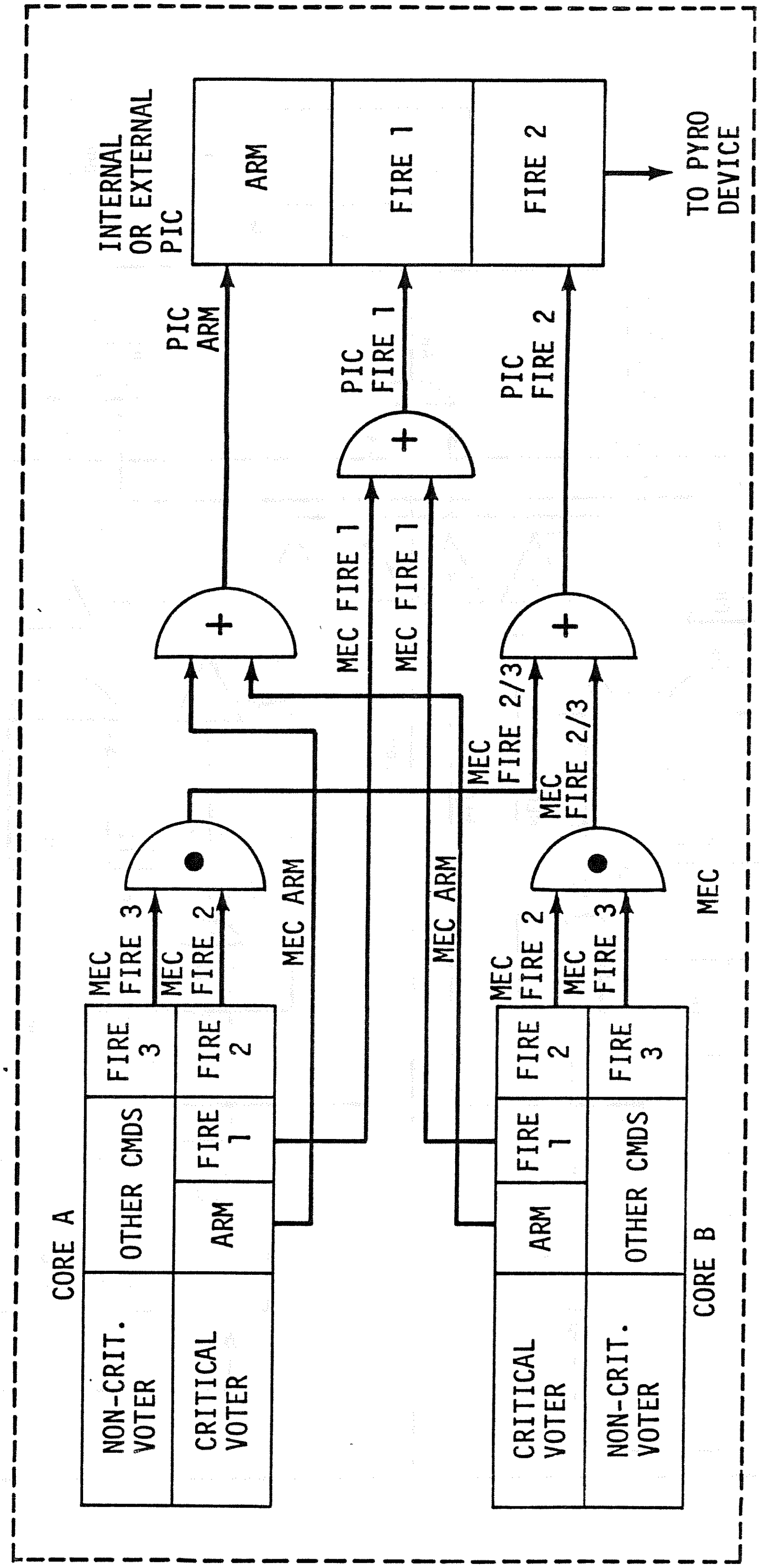


Figure 5.2-8.- Simplified diagram, voting interlock for PIC Fire 2/Fire 3 logic.

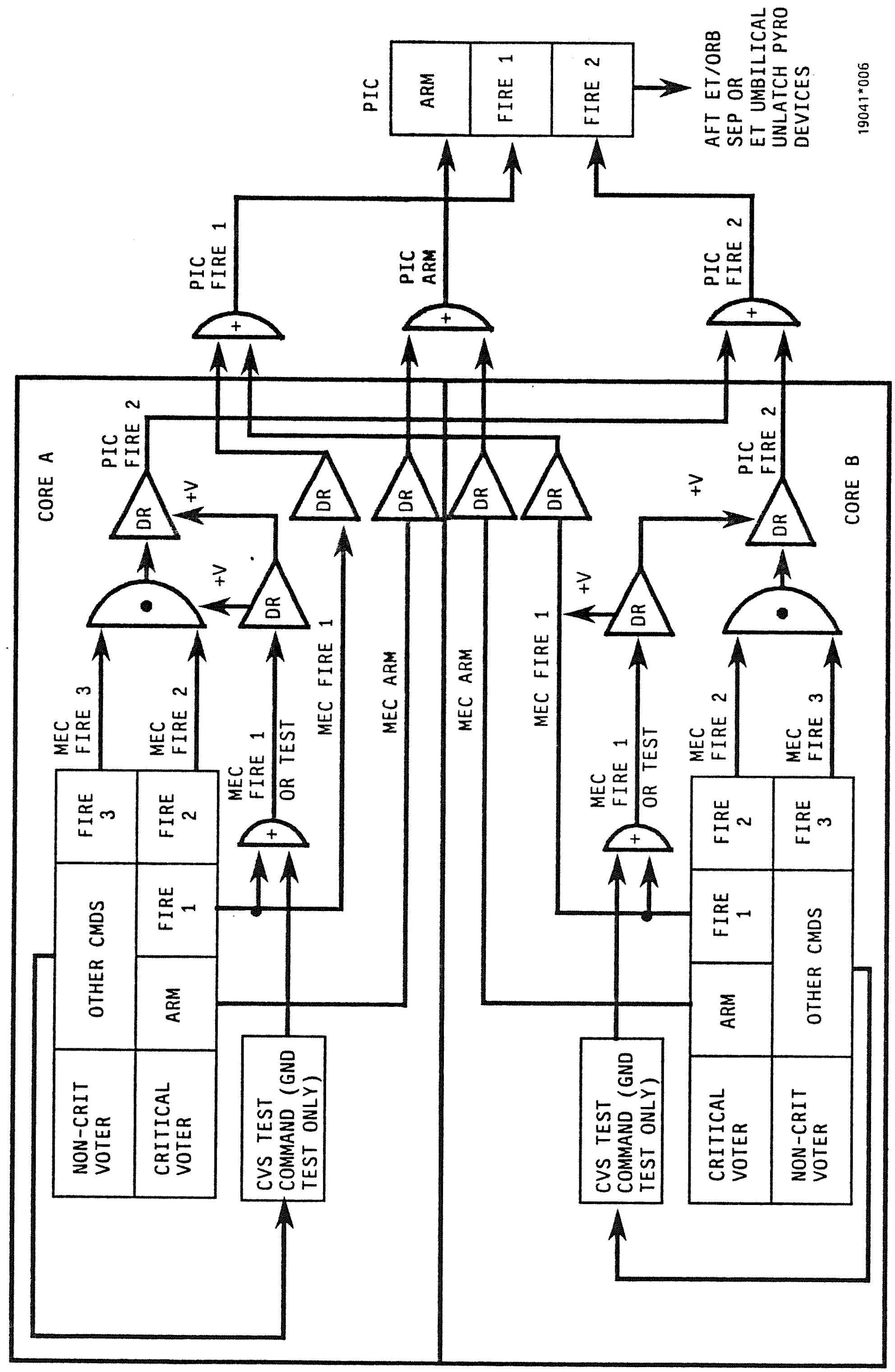


Figure 5.2-9.- Simplified diagram, ET/Orbiter separation and ET umbilical unlatch voting and interlock logic.

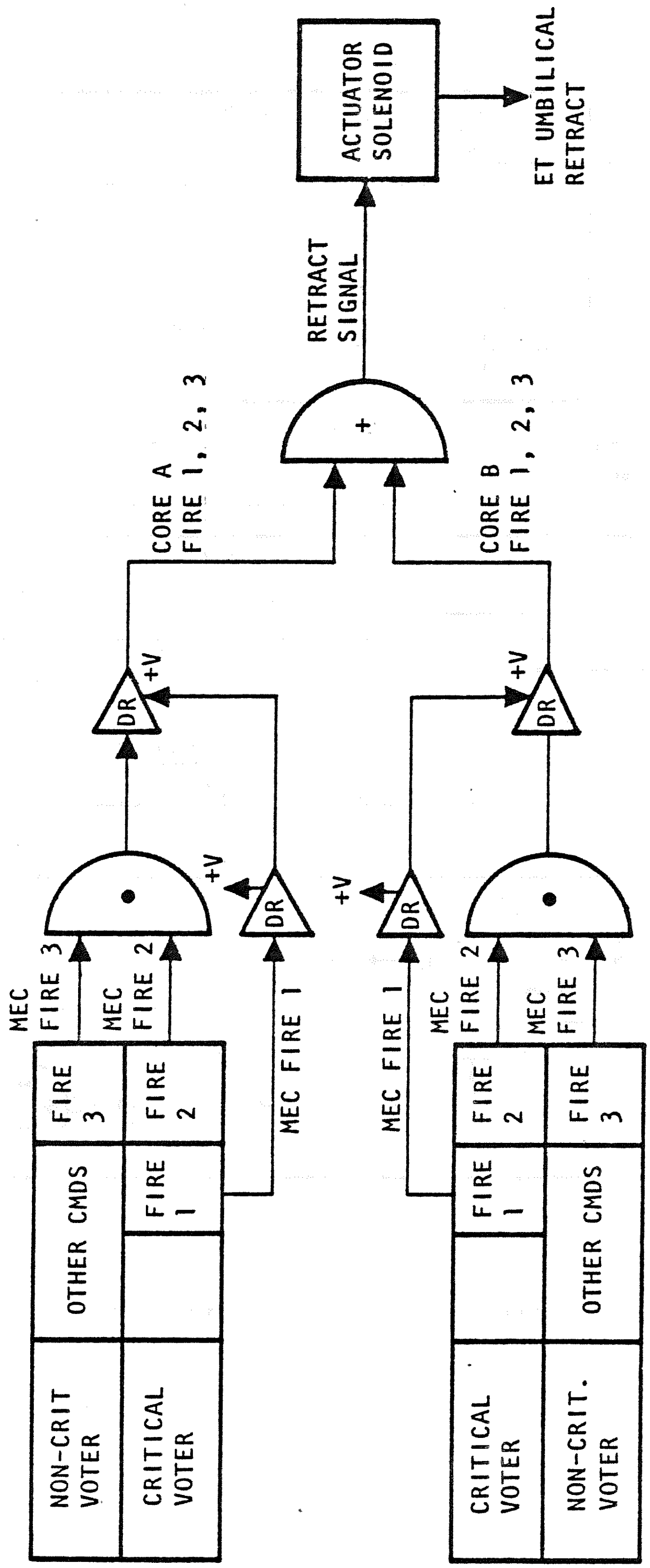


Figure 5.2-10.- Simplified diagram, ET umbilical retract voting and interlock logic.

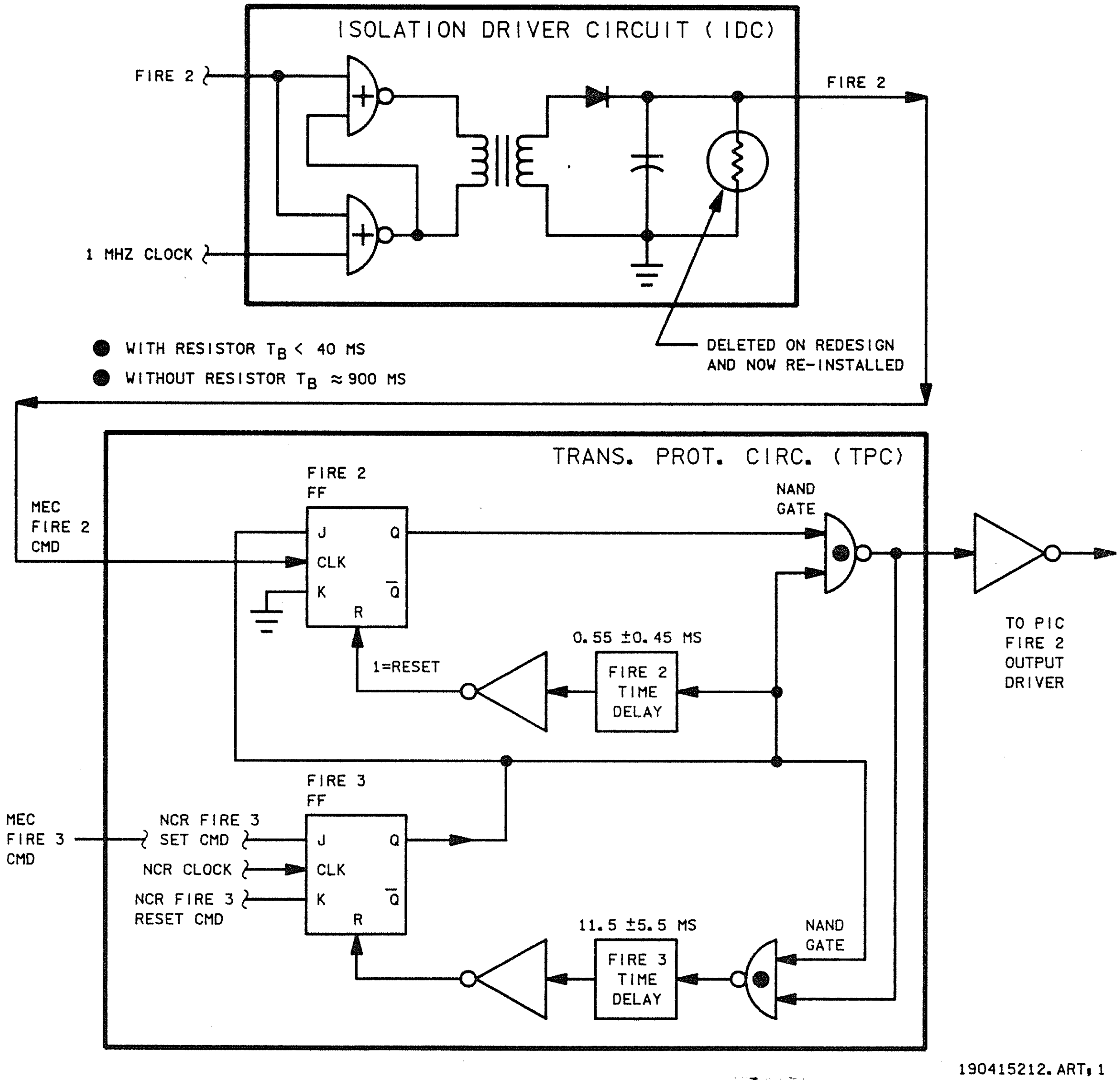


Figure 5.2-11.- The isolation driver circuit and the transient protection circuit.

5.3 ENHANCED MASTER EVENTS CONTROLLER OVERVIEW

5.3.1 Purpose

This brief provides a brief overview of the enhanced master events controller (EMEC) and insight into hardware differences between the EMEC and the MEC. The MEC is the interface unit between the GPC's and the SRB, the ET, and the orbiter. The MEC/EMEC receives commands from the GPC's and then compares and executes them during launch, SRB separation, and ET separation. The MEC/EMEC uses BITE to monitor the status of the pyrotechnic devices and the MEC/EMEC logic paths used to fire those devices. MEC/EMEC power can be controlled manually by the crew via the cockpit switches on panel 017.

The EMEC design meets the orbiter end item specification FAIL OPERATIONAL/ FAIL SAFE requirement for both premature and assured PIC firing. This requirement is not met for premature PIC firing. Critical PIC commands in the EMEC are channelized independently in each core to prevent any single or dual failure from causing premature execution of all commands.

The EMEC is designed to be compatible with the existing software and hardware interfaces, although it incorporates significant modifications to the MEC. The EMEC is more reliable (containing fewer parts) and implements improvements in operation, testing, and checkout.

5.3.2 MEC/EMEC Transparency Comparison

Functional transparency means the EMEC is physically and functionally interchangeable with the MEC, can be processed through the orbiter checkout system without modification to that system, and needs no interface modification to operate in the space shuttle environment.

Certain design criteria had to be addressed to ensure that control interfaces remained transparent between the MEC and the EMEC. Electrical input and output signal interfaces had to remain the same to ensure compatibility with existing MIA inputs and PIC outputs because discrete outputs will be electronically identical to the current MEC design. Mechanical and electrical hardware interfaces also had to remain the same. Table 5.3-I outlines a checklist of EMEC enhancements that have been made, while allowing the EMEC to be transparently interchanged with the MEC. Figure 5.3-1 depicts the EMEC in block diagram form. Figure 5.3-2 is a representation of the flow of EMEC commands to their respective PIC's. These command outputs are listed in table 5.3-II.

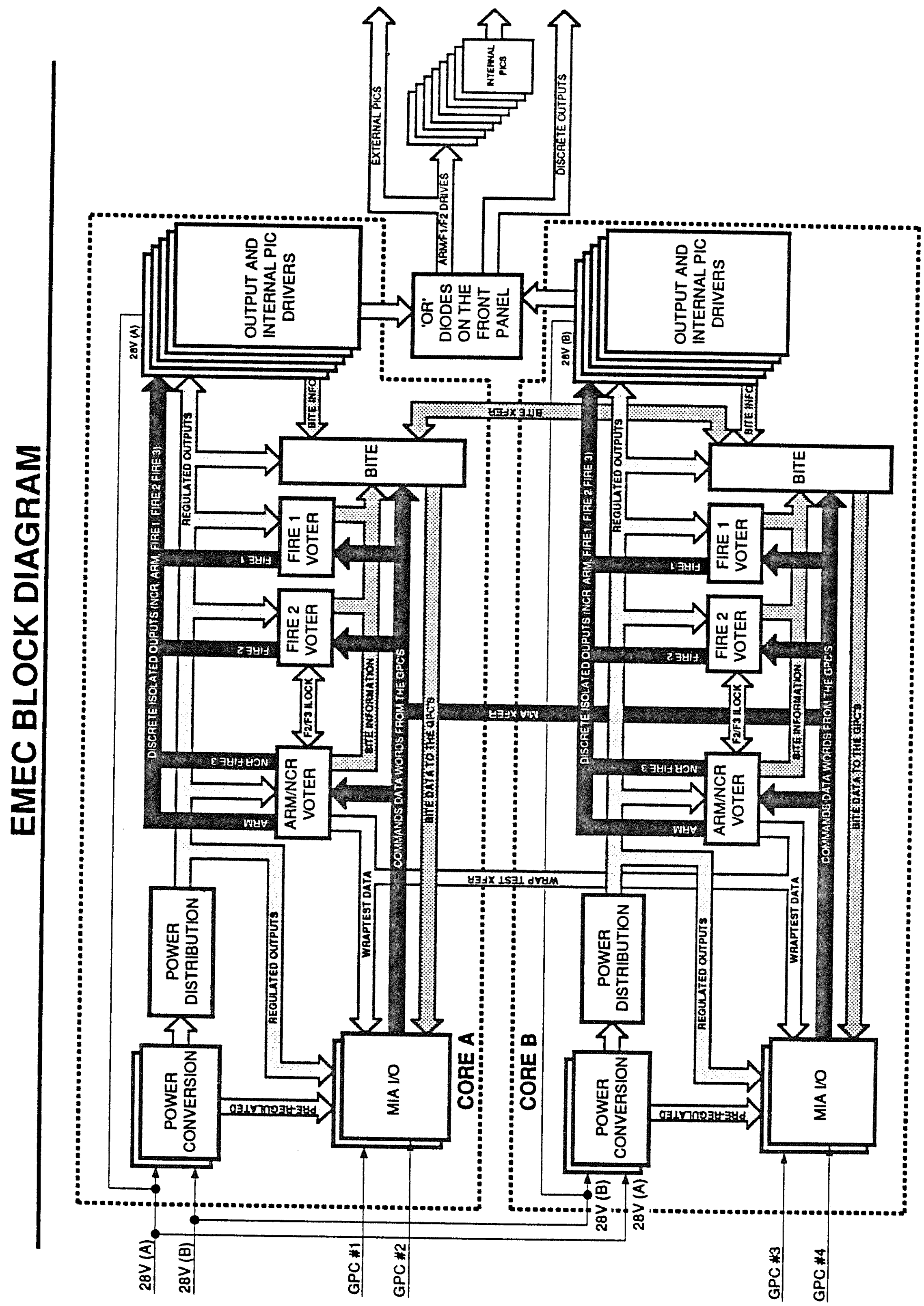


Figure 5.3-1.- EMEC block diagram.

TABLE 5.3-I.- CHECKLIST OF EMEC ENHANCEMENTS

Characteristic	Comments
Mass properties	Acceptable to reduce weight and alter the center of gravity
Footprint	EMEC makes use of existing mountholes
PIC's no longer require FIRE 3 commands	Results in an increase in fault tolerance
Power and thermal	Enhancements result in net decreases in both power and thermal transmission
Coldplate interface	The EMEC is framed rather than mounted on a flat surface, which eliminates bottom plate warping
Built-in test equipment	The MEC BITE is organized by a single subassembly whereas the EMEC BITE is organized by different subassemblies
Internal mechanical differences	The EMEC is completely different from a mechanical standpoint but is identical from an operational standpoint

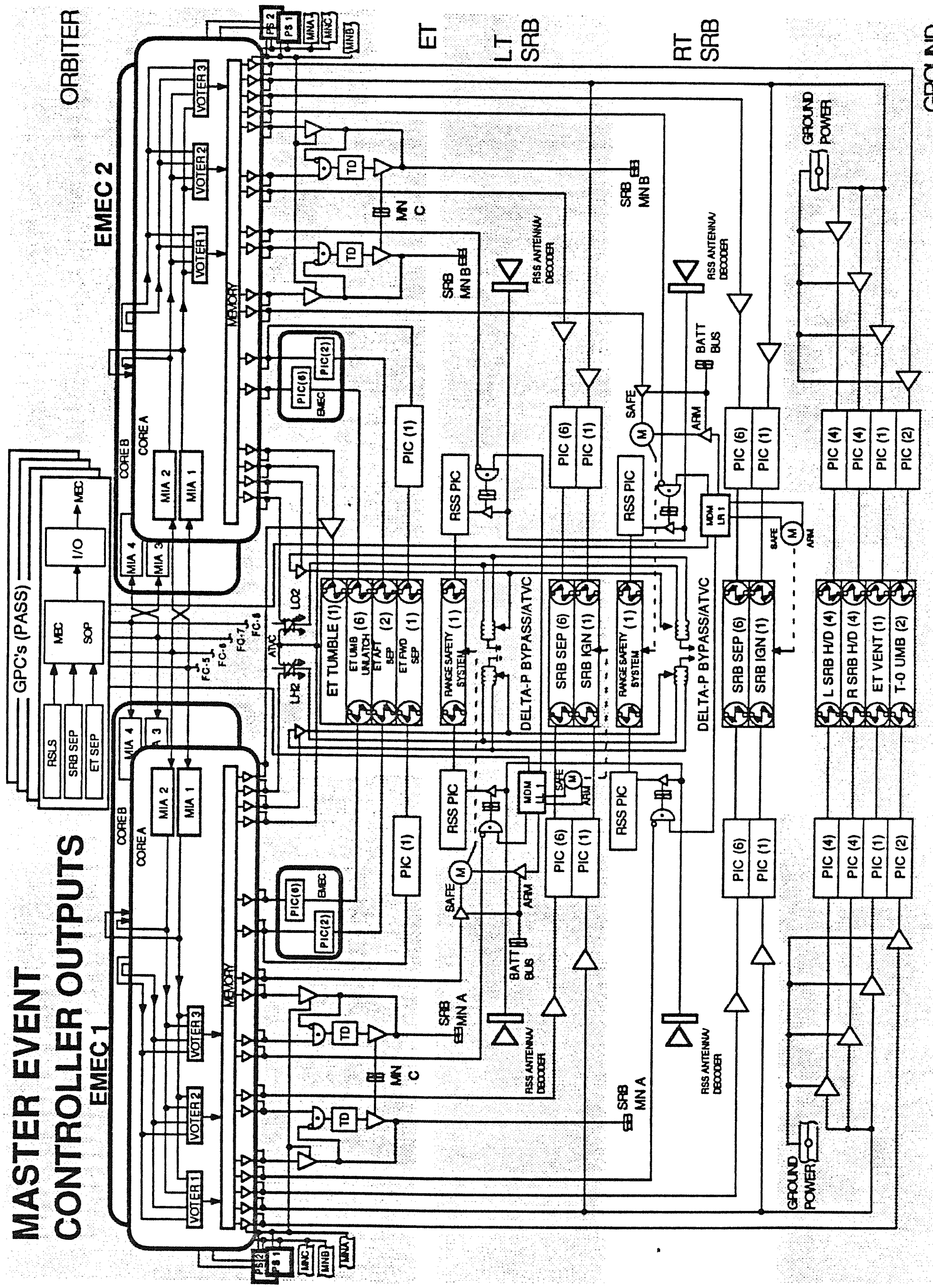


Figure 5.3-2.- Master events controller outputs.

TABLE 5.3-II.- EMEC OUTPUTS

Critical	Noncritical
All PIC functions	
• SRB ignition: ARM, FIRE1, FIRE2	SRB ignition FIRE3
• T-0 umbilical separation: ARM, FIRE1, FIRE2	T-0 umbilical separation FIRE3
• SRB separation: ARM, FIRE1, FIRE2	SRB separation FIRE3
• ET umbilical unlatch: ARM, FIRE1, FIRE2	SRB separation FIRE3
• ET/ORB structural separation: ARM, FIRE1, FIRE2	ET/ORB separation FIRE3
• ET umbilical retract: FIRE1, FIRE2	ET umbilical retract FIRE3
• ET tumble: ARM, FIRE	
• Ascent thrust vector control bias deadface 1 (2)	
• Ascent thrust vector control bias deadface 3 (4)	
• LT SRB range safety system safe 1 (2)	
• RT SRB range safety system safe 2 (1)	
	LT and RT SRB power
	LT and RT SRB backup power
	LT and RT SRB RSS power off
	Ascent thrust vector control isolation valve driver power 1 (2)
	Ascent thrust vector control isolation valve driver power 3 (4)
	ET instrumentation power

5.3.3 EMEC Command Handling

Noncritical Voting

Figure 5.3-3 is a block diagram of the EMEC noncritical voters. Noncritical command voting begins a NCR command word (CW) that is decoded by the MIA I/O module on one of the four channels. The logic path associated with NCR command voting is shown in figure 5.3-4. There are two voters for NCR commands in each core. One is for NCR set commands, and the other is for NCR reset commands. Either voter is enabled, depending upon the set or reset command. During the NCR register verification test, registers in both voters are enabled. The NCR CW is followed by the NCR command data word (CDW) that consists of 16 bits of data representing 16 NCR commands. This is in contrast to the CR CDW that has 16 bits of data representing one command. The CDW is validated and the appropriate 16 bits are sent to the NCR voter. The data bits are then shifted through a 16-bit shift register by the receive shift clock from the MIA. A validation bit for the data word is followed by the data bit. The validation bit enables the transfer of data from that channel to the comparison circuit (no validation bit is generated during the NCR register verification test).

A data word received on another channel is stored similarly in a 16-bit shift register. The validation bit enables the transfer of this data word into the comparison circuit, and the voting starts immediately. There is no skew time delay to allow for all four channel's data to be voted on. All six pairs are compared to obtain a two-of-four vote. These results are OR'ed together, such that any match will result in a set output. The exception to this rule is the resistance test command that can be set by just one channel (channel 1 for core A, channel 3 for core B). The set output is latched in a holding register and forwarded to the appropriate driver via optoisolators. The set output is fed back to clear that bit of the shift register to prevent the old data from being voted with the new incoming data. The reset commands are handled in a separate, identical circuit. The outputs of the reset voter are used to reset the holding register for the NCR set outputs.

There is a possibility of a two-on-two GPC set split condition where two channels attempt a set operation and the other two channels attempt a reset operation. In such cases, the channels that arrive late determine the mode (if both pairs were to come in simultaneously, a set operation would occur). Most commands are issued every 40 msec. Therefore, if an error occurs, it can be corrected in the next minor cycle of the GPC, provided that the error is not repeated during the next cycle.

A two-on-two set split could also occur in the data words where the two channel pairs have different data words. This first command to the EMEC during each pass of a minor cycle is the NCR set. Any matching pair of 1's in the corresponding bit positions will result in the setting of that command bit. As a result, additional NCR commands will be set after the set operation. This is not a concern because early execution of NCR commands does not impose any threats. The FIRE 3 commands are automatically reset if the associated FIRE 2 command does not arrive within a certain window.

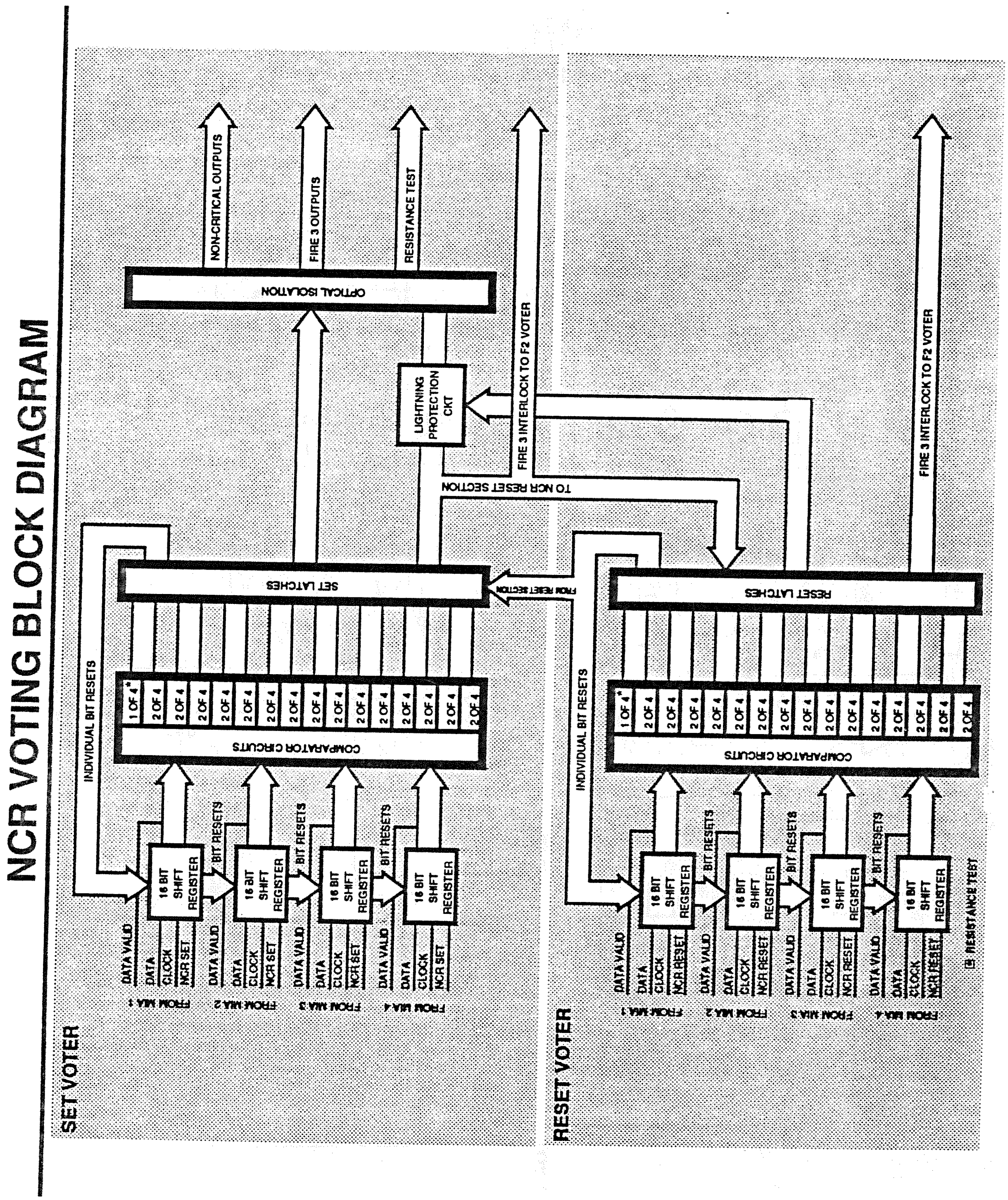


Figure 5.3-3.- NCR voting block diagram.

NON CRITICAL COMMAND VOTING LOGIC (1 BIT)

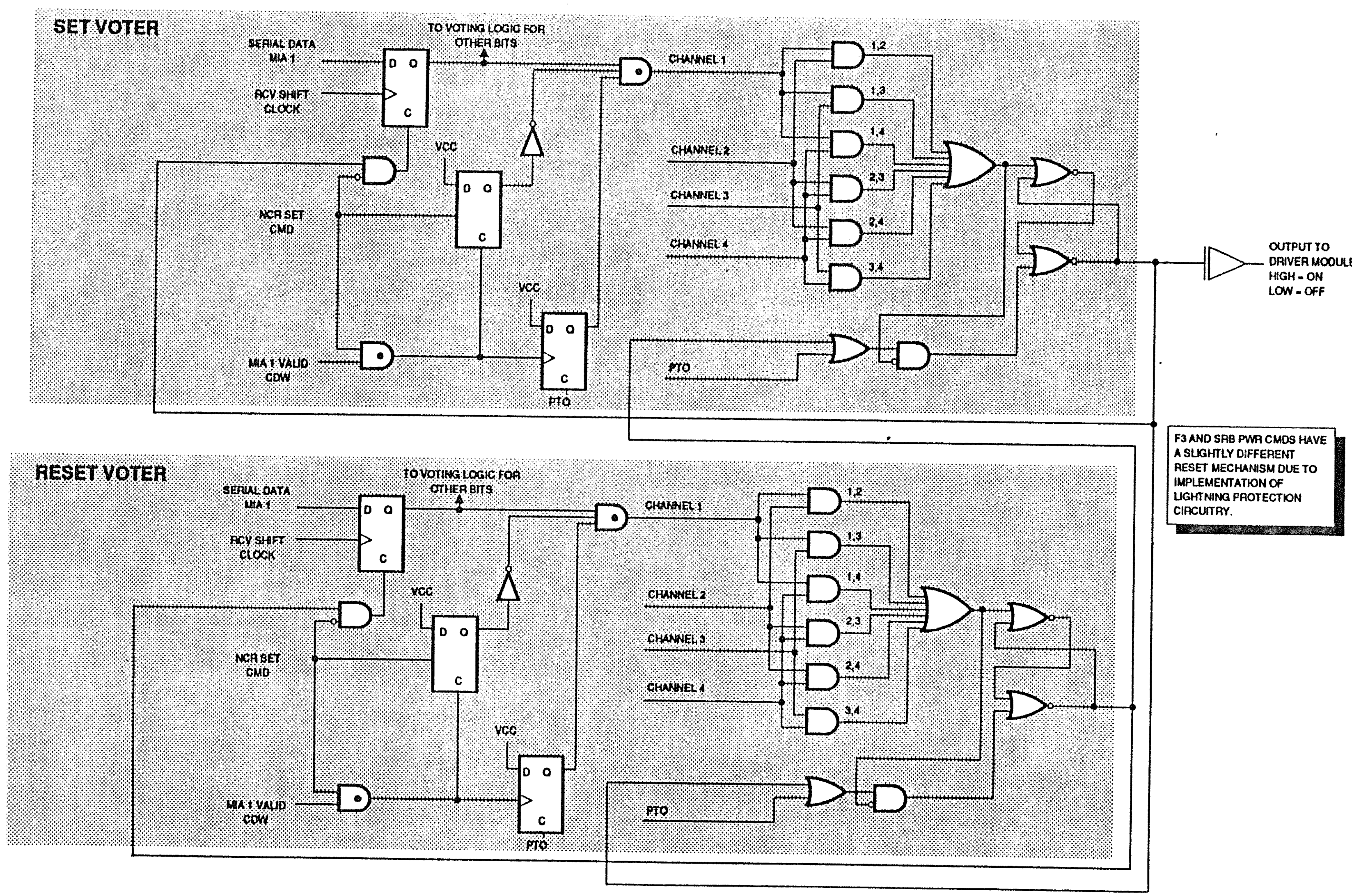


Figure 5.3-4.- NCR command voting logic.

About 40 µsec following the set message, the GPC's send a reset message whose data word is a complement of the set data word. This ensures that no uncommanded bits are in the set state. The reset data word for a channel is obtained by complementing the set data word for the same channel. Therefore, any errors in the set data word are reflected in the reset data word (GPC failure mode). Because reset voting is similar to set voting, all the bit positions that had a miscomparison between the two channels will be reset. The worst case occurs when the two discrepant data word pairs are complements of each other (assumes two smart failures). In that case, all NCR bits will be set after operation and reset approximately 640 µsec later. This action can prevent the FIRE 3 commands from being active for a sufficient time. However, the commands may not be active in the window necessary to set the associated FIRE 2 commands. This error can be corrected in the next minor cycle, provided the condition does not persist.

This situation is different from the MEC. None of the two failure modes mentioned above can occur in the MEC because it only votes two out of three channels for NCR logic. Therefore, there is no danger of a two-on-two split. The two matching channels will produce the appropriate outputs. If the third channel does not compare with the rest, it is discarded. In addition, output channels set by a core cannot be reset by the other core because the voter outputs of the two cores are OR'ed before setting the output flip-flop (1 indicates a set operation and 0 indicates a reset operation).

Additional Notes:

- Reset commands are handled in the same manner as the set commands, except for the manipulation of the holding register output of the set section.
- The existing MEC votes two-of-three for this function. The EMEC votes two-of-four for this function.
- No output is generated until the input is validated. Therefore, the resistance test cannot be set/reset without an NCR command (will not set during wrap test).
- FIRE 3 commands are also sent to the FIRE 2 section to provide lightning protection.

5.3.4 Critical Command Handling

Figure 5.3-5 is a block diagram of the EMEC critical voter logic. Critical command processing is initiated when a CR command is received and decoded by the MIA I/O module. The CW word is followed by four CDW's (it can accept less than four but both PASS and BFS use four). The 16 bits of each CDW from an MIA channel are clocked into a 64-bit shift register for data storage by the receive shift clock (1 Mhz). There are four data storage registers, one for each channel. The validation bit accompanying each data word is stored in a 4-bit shift register. This bit is shifted through the register by each new validation word.

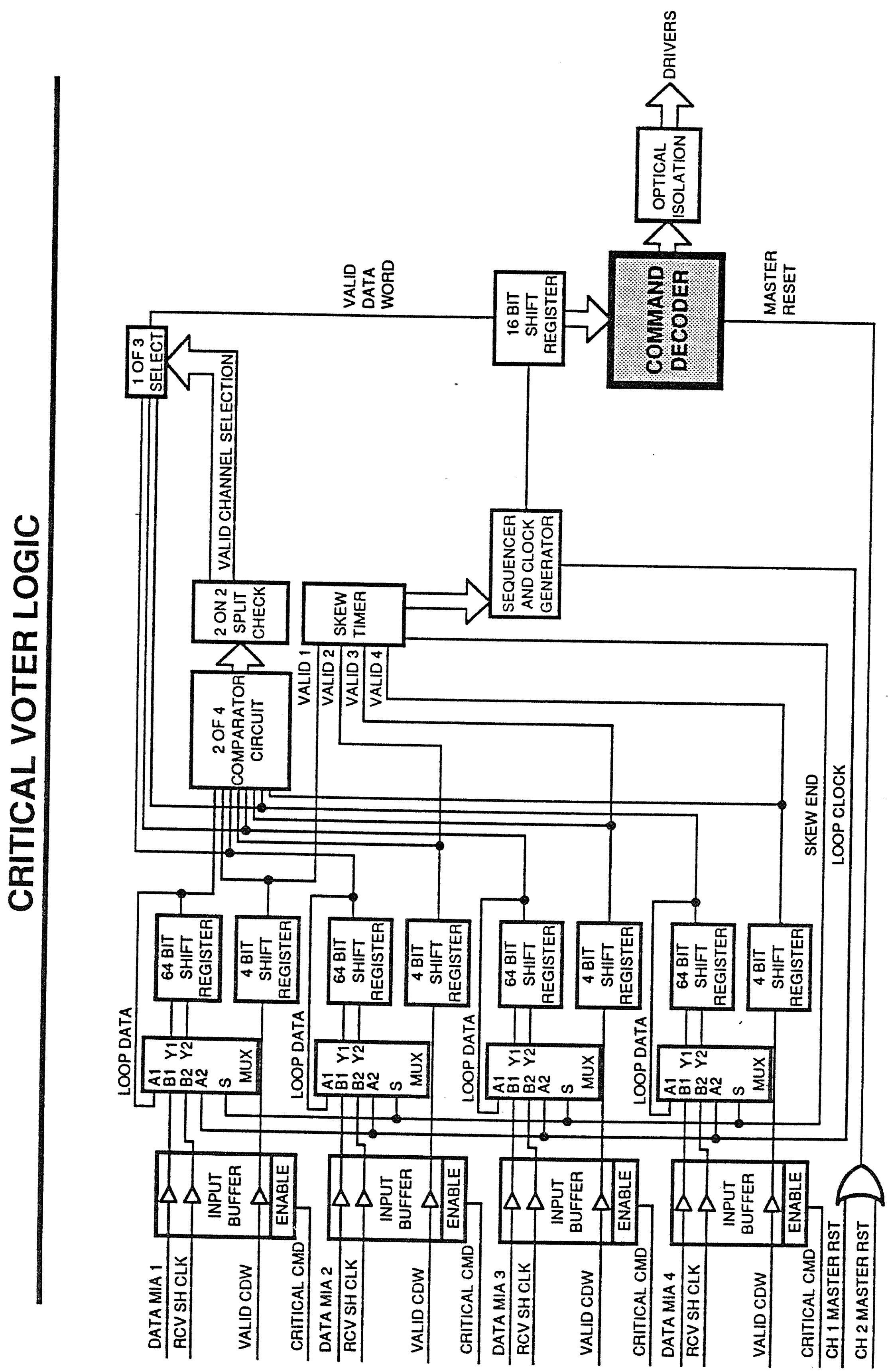


Figure 5.3-5.- Critical voter logic.

A skew timer is activated when the voter detects a critical CW on another MIA channel. Critical commands on any two channels activate the skew timers in all six critical voters (three in each core). The skew timer delays the comparison between the channels by 4.5 msec to allow for timing skew present between the outputs of the different GPC's. The delay allows all four channels to participate simultaneously in the voting process. The "skew end" signal goes high at the end of the skew period, indicating the initiation of the comparison mode. An internal clock (the loop clock) is generated by the voter shifting the data bits. The loop clock provides bursts of 16 pulses (1 μsec), with a gap of 16 μsec between the bursts. Each burst of the loop clock is used to move a data word from the 64-bit data storage register into the bit comparison circuit. The data bits are looped back into the register to be used again for command decoding as they are moved out of the storage register. The data are not retained by the comparison circuit. Therefore, the data pass through the 64-bit storage register twice during the voting process.

A bit-by-bit comparison is performed among all six pairs of data from the four channels (1,2; 1,3; 1,4; 2,3; 2,4; 3,4) as the 16 bits of data pass through the comparison circuit. A miscompare among any channel pair sets the miscompare flip-flop for that channel pair. The "comp" pulse, appearing about 9 μsec after the 16-pulse burst of the loop clock, clears the miscompare flip-flop for the next word. The validation bit for each word is shifted out of the 4-bit shift register by the "valid" clock pulse, which occurs immediately after the comp pulse. If either of the two channels being compared does not have the set validation bit, a miscomparison is declared.

The voters perform two-of-four voting; i.e., if two of the four channels have valid and identical data for a particular CDW position, that data word is selected to be forwarded for command decoding. If there are two pairs of data that match within the set, but the two pairs do not match, a two-on-two split is declared and none of the two commands is decoded (except if one of the sets is all zeros). The logic required to perform the two-on-two critical comparison check is depicted in figure 5.3-6. Nominal voting results in a selection of one of the four channels for each word whose data are to be used for command decoding. Two 4-bit shift registers are used to store the voting results. The comp pulse is used for shifting the results through the register into a selection multiplexer, which selects data from the four input channels. These data are the recycled data from the 64-bit shift registers because the data for the first pass were lost in the comparison circuit. The data were looped back into the storage registers for this reason. The loop clock shifts the data into the selection multiplexer following the data comparison. The output of the multiplexer containing the correct data is fed into the 16-bit shift register that converts the data from serial to parallel.

The 16 bits of parallel data are then transferred into a decoder and checked to determine if they represent a legal command. The "cexec" pulse strobes each data word into the decoder. Critical command decoding logic is presented in figure 5.3-7. Upon a match, that command is passed on to the driver module to provide a 28 V dc signal discrete to the appropriate

CRITICAL COMPARISON/2 ON 2 SPLIT CHECK

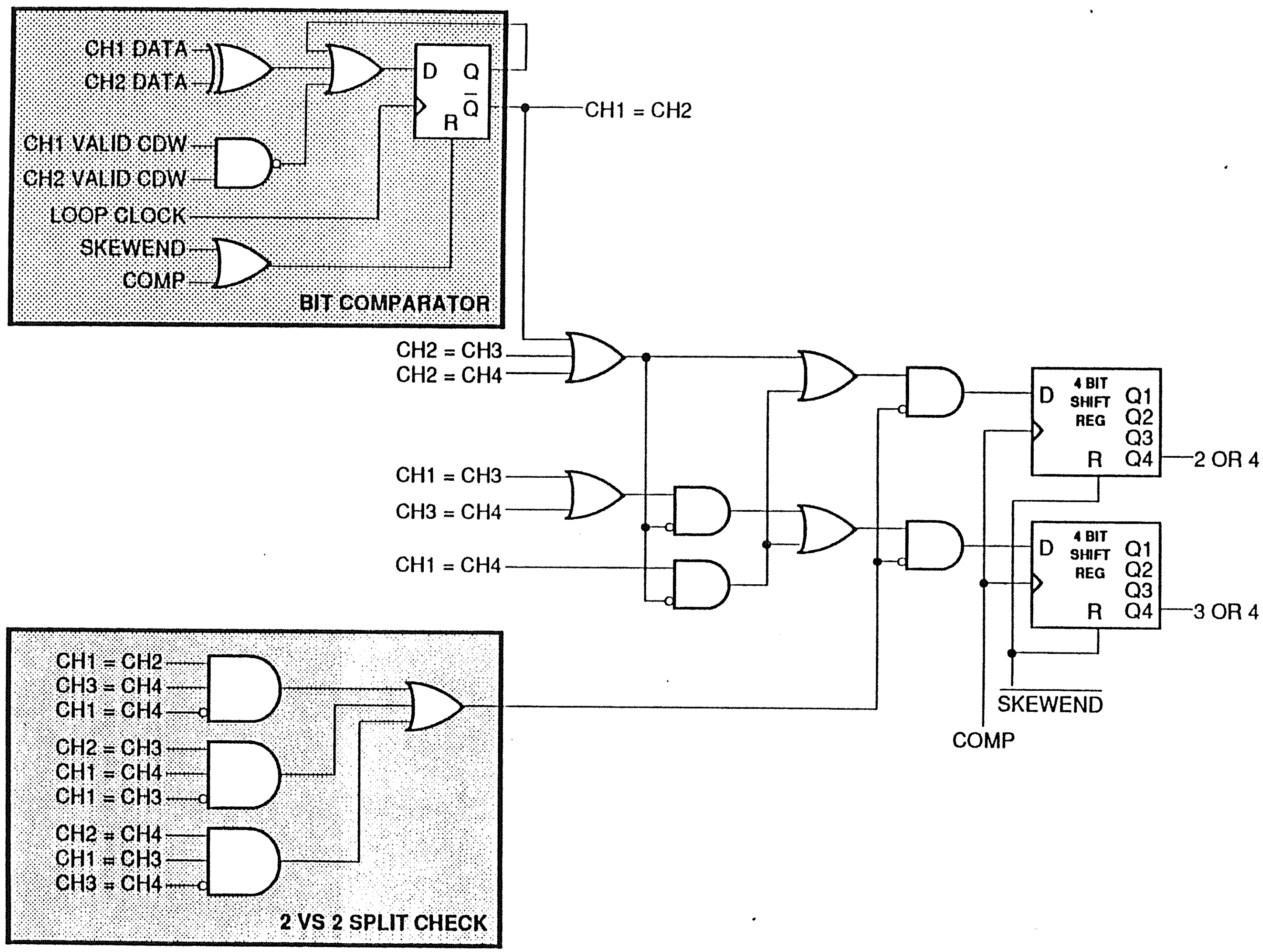


Figure 5.3-6.- Critical comparison/two-on-two split check.

CRITICAL DECODING

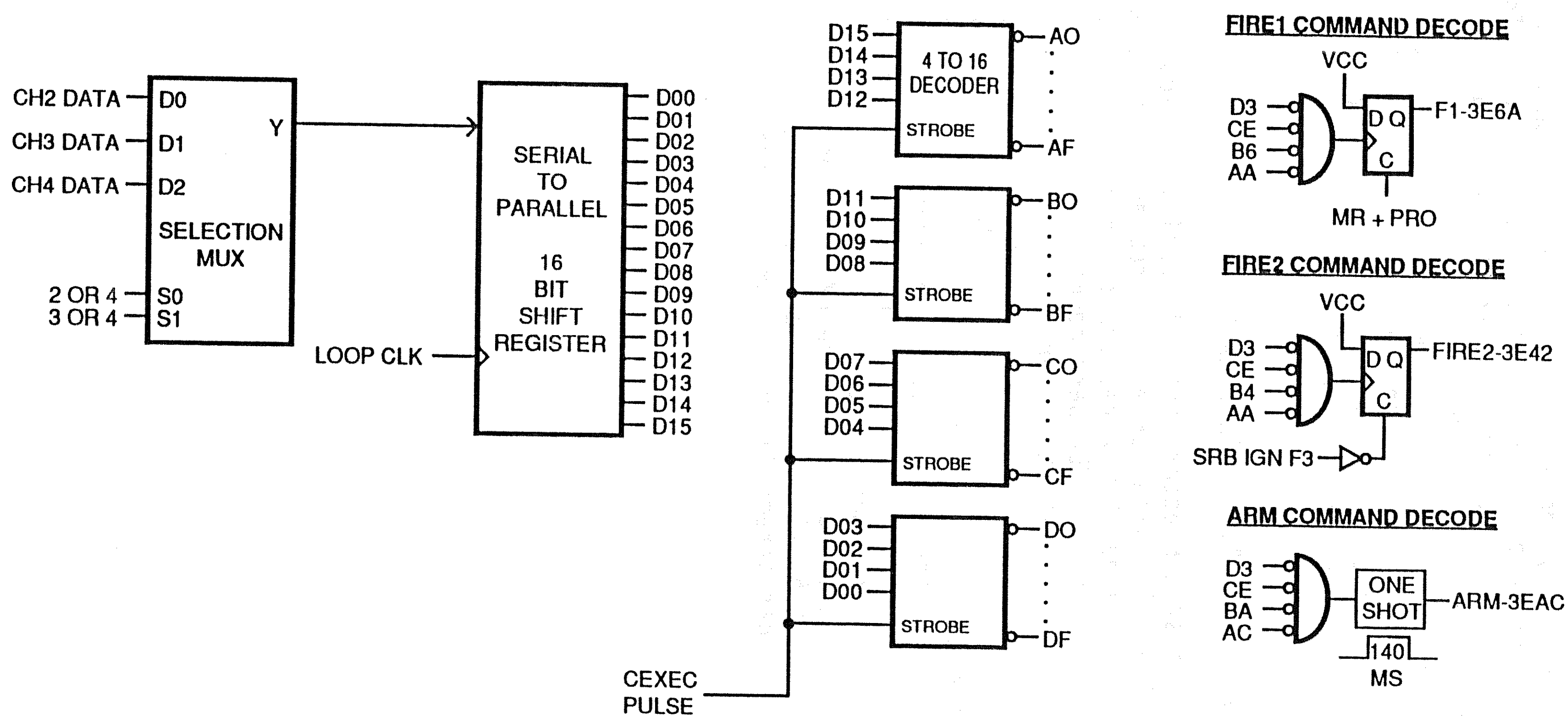


Figure 5.3-7.- Critical decoding.

interface. All three CR voters in a core perform the comparison and channel selection. However, only the ARM voter decoder can output ARM commands and only the FIRE 1 voter decoder can output FIRE 1 commands. The ARM command triggers a one-shot pulse that provides an output for 140 msec. It is normally retriggered before it times out to extend the PIC charge time (spec value is 1 sec minimum). The FIRE 1 and FIRE 2 commands set a latch. Command handling is identical for the ARM, FIRE 1, and FIRE 2 commands. ARM and FIRE 1 commands can be reset by the master reset command from the GPC, but the FIRE 2 command requires an NCR FIRE 3 reset command to be reset.

5.3.5 Mechanical Enhancements to the EMEC

The addition of a test connector port will facilitate additional ATP testing and fault isolation. The wire harness was replaced with a master interconnect board that eliminated 1500 wires and the need for handwiring the MEC. EMEC enhancements reduced the MEC part count by approximately 75 percent (7328 to 1900), which improves the thermal and vibrational characteristics of the MEC. The EMEC was partitioned into eight subsections that allow for additional fault isolation by function. The EMEC chassis was electron-beam-welded for additional strength and support. The EMEC module size was reduced by 60 percent, the module types were reduced from 9 to 7, and external module heat sinks were eliminated. The net result of these design improvements is the improvement of EMEC producibility, maintainability, and reliability. A design comparison of the EMEC to the MEC is depicted in table 5.3-III. Tables 5.3-IV and 5.3-V are representations of EMEC/MEC design differences that affect flight and ground operations, respectively.

TABLE 5.3-III.- EMEC/MEC DESIGN COMPARISON

MEC	EMEC
Two subcore design (core A/core B) Critical commands handled by one section per core - FIRE 3 commands handled in this section NCR commands handled by one section per core - FIRE 3 commands handled in this section	Two subcore design • Each subcore has eight modules • Each module has multiple identical subcores • Function required is picked off of a generic module • Two ARM, FIRE 1/FIRE 2 subsections • Two NCR/FIRE 3 subsections. No longer needed for qualification of critical commands Generic module design allows for fault tolerant NCR command
Use of SSI complexity logic (CMOS 4000 series) • Required high-density (10-16) layer board • Logic density of about 100 gates/in^2	Use of LSI complexity logic* • Requires medium-density (4-8) layer board • Logic density of about 1000 gates/in^2

* LSI complexity logic implements new technology to design integrated circuitry logic. This new technology allows for increased computational speed obtained with lower power consumption. Logic functions are implemented using electrical programmable logic devices (EPLD's) instead of metal oxide semiconductors (MOS) standard logic elements. EPLD's are logic devices that perform in a manner complementary to the MOS chips but consume less power and operate more efficiently. Logic gates are designed by being burnt into the EPLD devices.

TABLE 5.3-IV.- EMEC VERSUS MEC DIFFERENCES AFFECTING FLIGHT OPERATIONS

Critical command voting • Both the EMEC and the MEC perform critical functions, based on a two-of-four vote (provided there is not a set split). In the case of the set split, a no-match position is selected • The EMEC starts a skew timer on receipt of a second command, regardless of the channel • The MEC will start a skew timer on channel 1 or 2 for core A, and channel 3 or 4 for core B
Noncritical command voting • The EMEC will perform NCR functions based on a two-of-four vote • The EMEC will execute NCR commands in the case of the two-on-two set split • The MEC will perform NCR functions based on a two-of-three vote
ET tumble valve • The EMEC qualifies the ET tumble valve arm command with the fire command (ref. fig. 5.3-1) • The MEC does not qualify the ET tumble valve arm command
Solenoid and arm driver wave shapes • Rise and fall times are different but meet specification requirements • The shape of the rising edge is "RC" rather than linear • This design improvement is associated with the elimination of an FMEA noncompliance

TABLE 5.3-V.- MEC ENHANCEMENTS AFFECTING GROUND OPERATIONS

Operational BITE* (OPB) • EMEC command channel bits are only reset by a power cycle • MEC command channel bits are reset by most BITE commands • BITE response will differ between EMEC/MEC, depending on the previous command sequence • Operational BITE is only used for troubleshooting
Wrap test • The EMEC returns an all-zero pattern (no voltage) in return data word (RTD) 5 for any unused PIC channels • The MEC returns a simulated voltage in RDW 5 for any unused PIC channels
Reset of critical commands • The EMEC will set CR commands in both cores based on a two-of-four channel vote • The MEC requires channel 1 or 2 to set core A and channel 3 or 4 to set core B
Preflight BITE (PFB) • Nominally, the PFB response will be identical for the MEC and the EMEC • The MEC requires first-pass cleansing to the PFB word prior to obtaining accurate data, following power up • The EMEC provides accurate data on the first request

5.3.6 Special Feature of the EMEC (Isolation/Lightning)

FIRE 2/FIRE 3 Logic Interlock

The FIRE 3 command was added to the PIC firing sequence of ARM, FIRE 1, and FIRE 2 to provide protection against premature firing of the PIC because of an overvoltage failure mode of the power supply common to the critical voters. The FIRE 3 command generated in the noncritical voter may cause the ARM and FIRE 1 commands to be on, but absence of FIRE 3 will keep the FIRE 2 off, preventing a premature PIC firing.

The purpose of the timing circuit associated with the FIRE 2 and FIRE 3 flip-flops, as shown in figure 5.3-8, is to inhibit an inadvertent setting of the FIRE 2 command by a single or multiple lightning strikes. The assumptions are that the duration of a lightning pulse will not exceed 50 μsec and that the lightning restrikes occur not less than 17 msec apart from each other.

The FIRE 3 command has to be set, which removes the FIRE 2 flip-flop from the reset state, thereby setting the FIRE 2 flip-flop. The rising edge of the FIRE 2 signal will latch the FIRE 2 command. Upon receiving a FIRE 3 command from the noncritical voter, two flip-flops delay the FIRE 3 command. The flip-flops operate on a 64 μsec period clock. Two rising edges of the clock are required for a HIGH to appear at the output of the FIRE 3 delay flip-flops, which removes the reset on the FIRE 2 flip flop. Therefore, if a lightning pulse activates both FIRE 2 and FIRE 3 signals, the FIRE 2 flip-flop will stay in a reset state for at least 64 μsec (128 μsec max.), which is greater than the 50 μsec duration of the lightning pulse.

The FIRE 2 flip-flop reset is removed, leaving it vulnerable for a restrike after 128 μsec (max.). To provide protection against a restrike setting the flip-flop, the FIRE 3 command is removed within 8.192 to 16.384 msec after the initial FIRE 3 arrives, provided no FIRE 2 command has been received during that time. (In the MEC, the FIRE 3 flip-flop resets in 6 to 27 msec.) The lightning protection circuit provides complete protection against defined lightning threats because a lightning strike does not occur until at least 17 msec after the first strike (according to the requirements in place during the EMEC design).

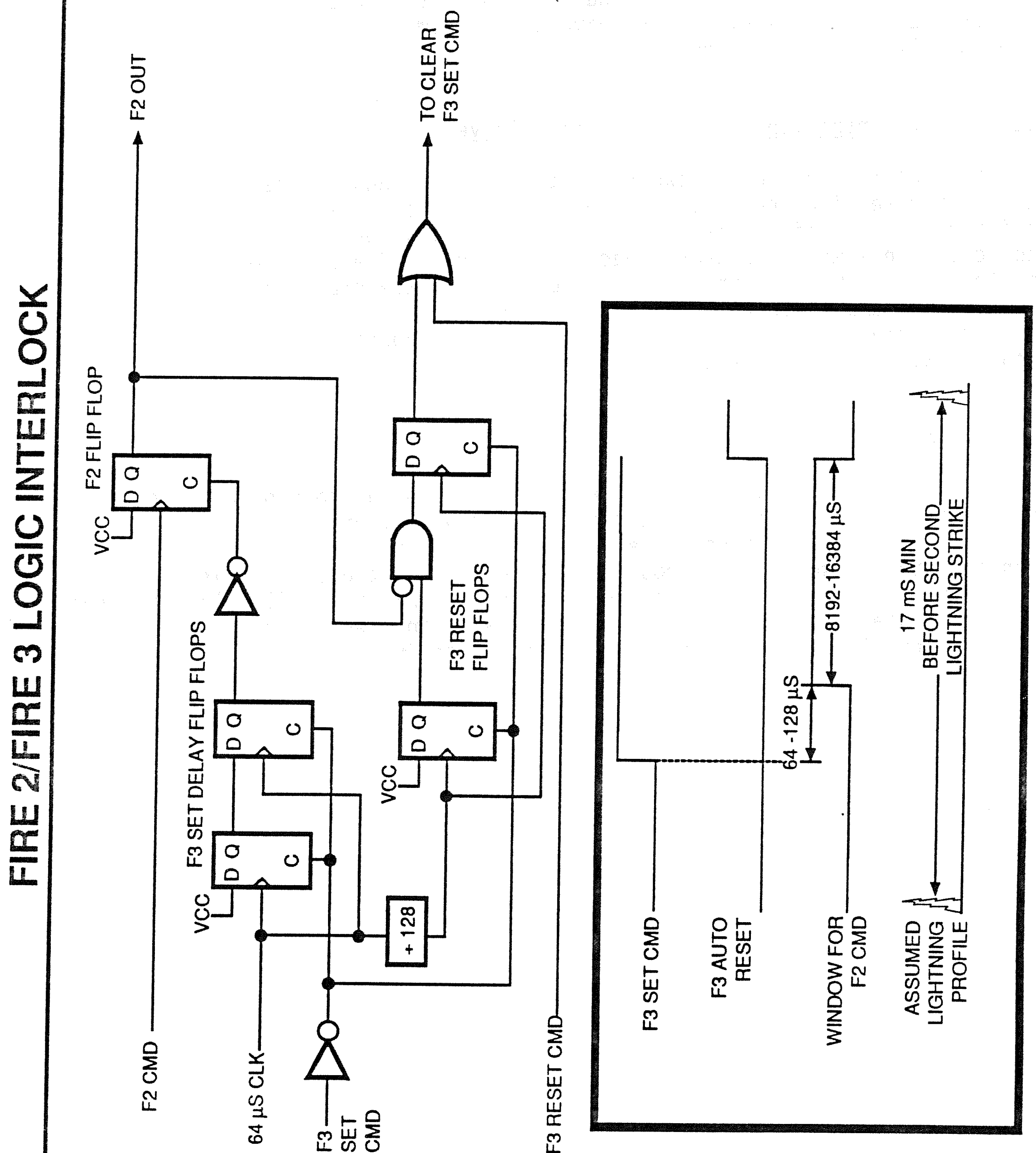


Figure 5.3-8.- FIRE 2/FIRE 3 logic interlock.

5.3.7 MIA Details

The MIA can transmit BITE data back to the GPC's upon request in addition to its receive functions. The MIA module performs all data formatting as a post-processing function. The receive and transmit sections of the EMEC are independent; the BITE may be polled at any time.

5.3.8 EMEC Enhancements to the Solenoid Driver

The EMEC solenoid driver is two-fault tolerant instead of one-fault tolerant as is the case with the MEC. Darlington transistors are replaced with low-resistance field effect transistors. Complex ramp generator networks are replaced with simple resistor/capacitor networks. The enhanced EMEC solenoid drivers eliminated the resistor select-in-place procedure by using a differential amplifier with one input of the amplifier connected across a voltage reference, which set current limiting at approximately 3 amps. The enhanced solenoid drivers also eliminated custom inductors.

5.3.9 Summary

The EMEC incorporates up-to-date electrical technology to provide a more efficient, compact, and reliable piece of hardware than the existing MEC. The EMEC is easier to maintain because it has fewer parts and was designed such that troubleshooting is modular. As a result, troubleshooting can track a particular failure to a specific area within the EMEC. The EMEC can be incorporated without impact to flight operations, while ground operations are more efficient because of enhancements in the BITE.

6.1 HELIUM SIGNATURE TEST

This systems brief is written to describe how the mps helium signature test is performed.

6.1.1 Introduction

The helium signature test is performed as a final check out of the mps propellant/pressurization systems. The purpose of the test is to verify that all parent metal and welds in the system are in working order. The test is broken down into the four identical subtests. These subtests verify the integrity of gaseous hydrogen, gaseous oxygen, liquid hydrogen and liquid oxygen systems. Each of the subtests calls for its system to be pressurized with helium and then gas samples to be taken from the aft compartment. From the gas samples a leak rate is calculated and compared to the known leakage of the system. If for some reason the leak is greater than expected the system would be considered suspect and further troubleshooting would be performed prior to a launch attempt.

6.1.2 Test Set Up

Before performing a helium signature test the mass spectrometer is calibrated. This is done by introducing three known concentrations of helium into the mass spectrometer and then calibrating the machine to read the known concentrations. While the mass spectrometer is being calibrated, the aft compartment is also being prepared for testing (fig. 6.1-1). This is done by sealing the aft compartment and introducing a 1370 scims purge rate (fig. 6.1-2). This purge rate maintains a positive delta pressure across the aft compartment to ensure that mixing of all gases occurs. Once the proper delta p is established, helium is injected into the aft compartment in three different areas. These areas are the LH_2 umbilical plate, the base of the heat shields, and the SSME hot gas system. At each concentration level, the purge is continued until the mass spectrometer reading is stabilized. Mass spectrometer readings are then taken to understand the mass spectrometer parts per million (ppm) readings and the known helium purge. These data are then used to generate a plot of helium purge rate in scims vs. mass spectrometer reading (fig. 6.1-3). After the linear curve is established, two uncertainty curves are drawn. These two curves represent a ±1 ppm reading to the actual. After this baseline is established the helium injection is terminated and the aft is verified clean of all helium. At this point the test preparations are complete and leak testing can begin.

6.1.3 Testing

Once the calibration of the instrumentation has occurred, one of the four systems is pressurized to 40 psia. The system is kept at pressure until a stabilized reading occurs on the mass spectrometer. After stabilization, the data point is taken and plotted on the ppm vs. scim leakage graph. This value is compared to the maximum allowable leak rate (9 scim). If the leakage rate was above this value the system would be considered suspect and the troubleshooting would be performed prior to the mission. After one system is cleared, the helium is vented and the aft compartment helium concentration is allowed to go to zero. Once this has happened the test procedure is repeated until each of the four systems is checked.

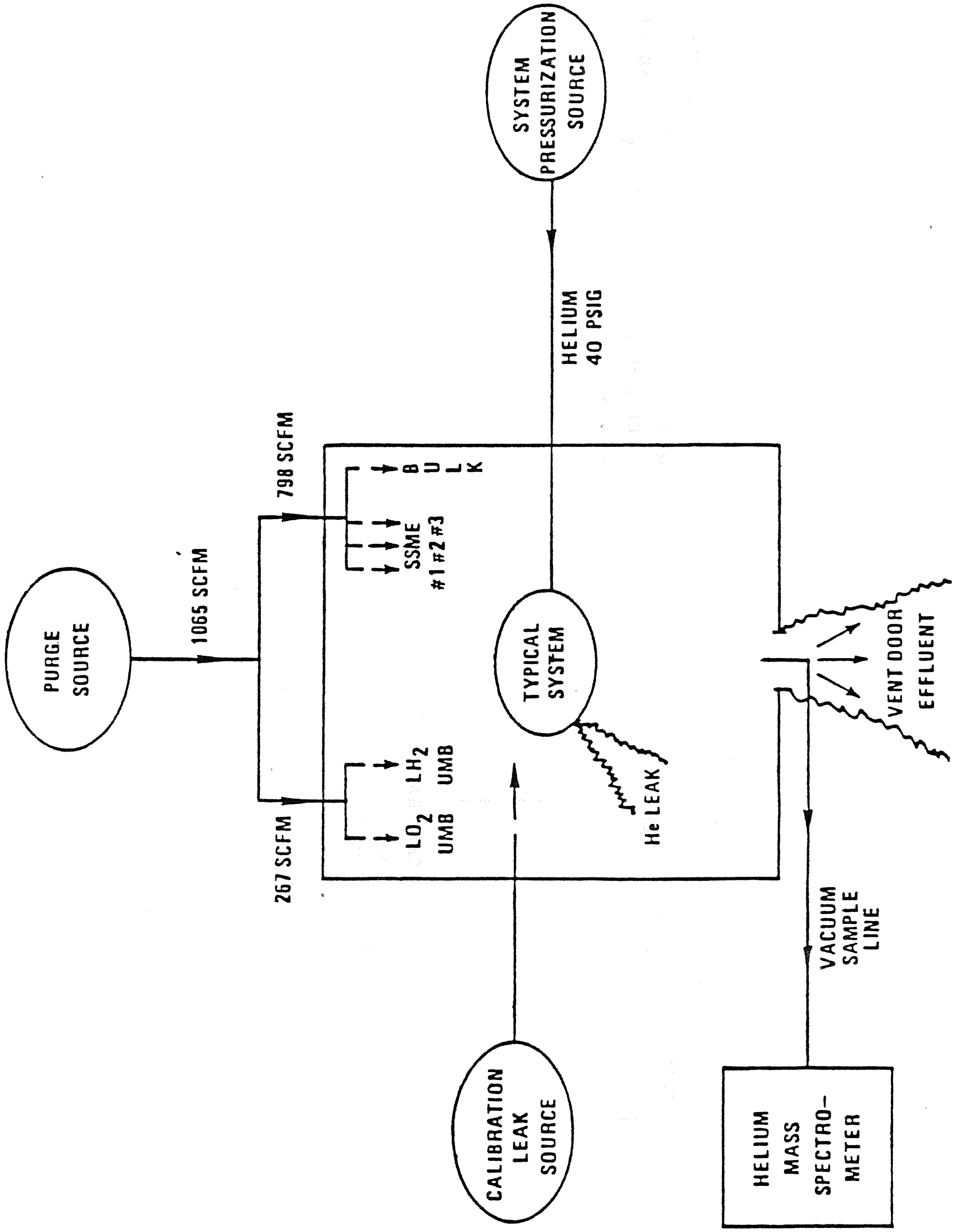


Figure 6-1.- Schematic of helium signature test setup.

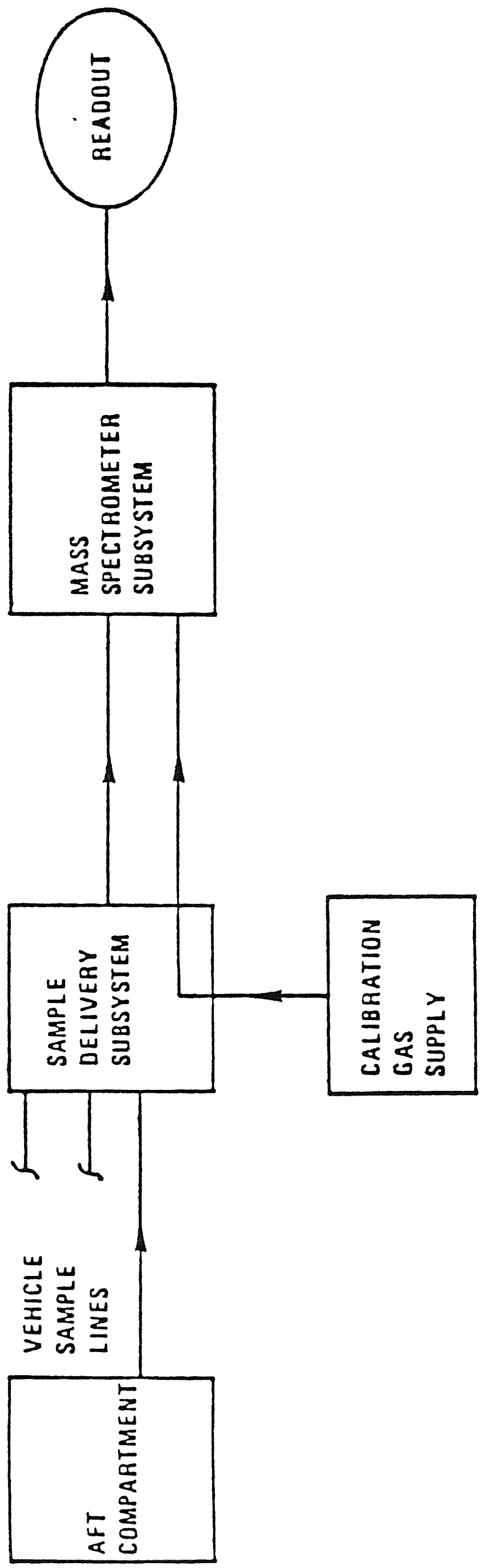


Figure 6-2.- Ground hazardous gas detection system (HGDS) schematic.

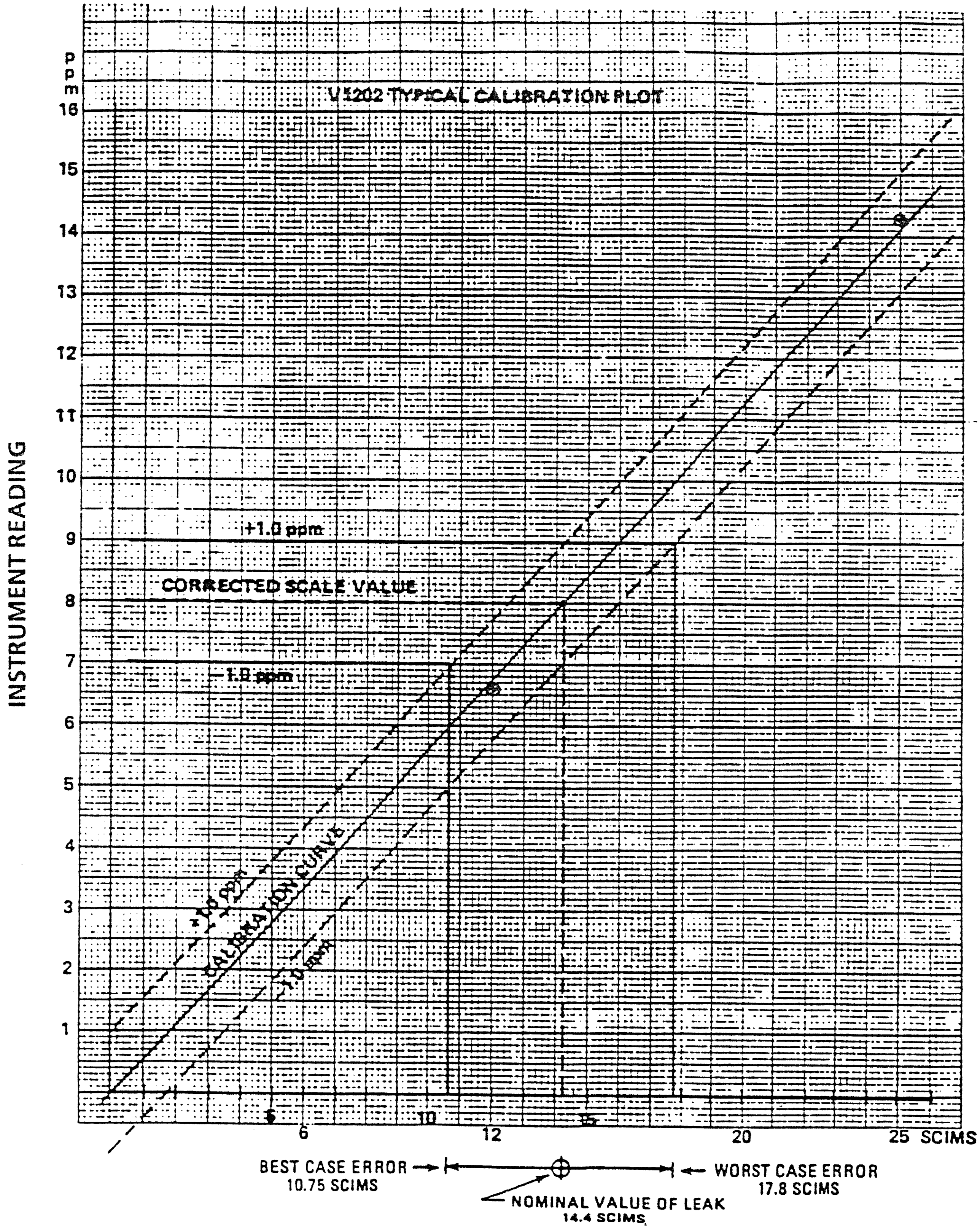


Figure 6-3.- Helium purge rate vs. mass spectrometer.

6.2 SWITCH REDUNDANCY MANAGEMENT

6.2.1 General

Control for a typical MPS valve is shown in figure 6.2-1. The LH_2 OUTBOARD FILL/DRAIN valve is controlled by the Sequencing Software in the PASS or BFS GPS's. The commands to open or close the valve come through FA MDM's. The crew can override the software command by using the cockpit switches. This switch has only one contact for the open command and one contact for the close command. However, some MPS switches have more than one contact for each position. The RTLS OUTBOARD DUMP VALVE is shown in figure 6.2-2. The valve, like the LH_2 OB F/D vlv is also controlled by the sequencing software. This valve also has crew override capability, but there are two contacts available for the open command and two contacts available for the close command for redundancy. The status of each contact is fed to the GPC's via FF MDM's. In the PASS, a special subroutine monitors the switch contacts and performs voting logic. The results of the voting logic in terms of switch position are then sent to the sequencing software. This systems brief discusses how the redundancy management voting logic works for MPS switches. The differences between the PASS and BFS are also discussed.

6.2.2 Selection Filters

Each available contact on the switch goes through a filter. If the redundancy management is monitoring a switch with a three contacts per switch position, a two-out-of-three majority vote is used to determine the switch position. If the switch has two contacts per switch position, the selection filter is the "logical and" of the available contacts. That is both A and B contacts must be true (logical 1) for the selection filter output of switch position to be true. If only one contact is available, the selection filter is a simple one-of-one vote. See table 6.2-I. A switch contact is available if the associated FF MDM is not commfaulted. If a three-contact switch has one contact commfaulted, the selection filter downmodes to a two-contact switch selection filter. The voting logic goes from two-of-three to "logical and." If a two-contact switch has one contact commfaulted (or a three-contact switch has two contacts commfaulted), the selection filter downmodes to a one-contact selection filter. The "logical and" voting logic becomes a one-of-one vote.

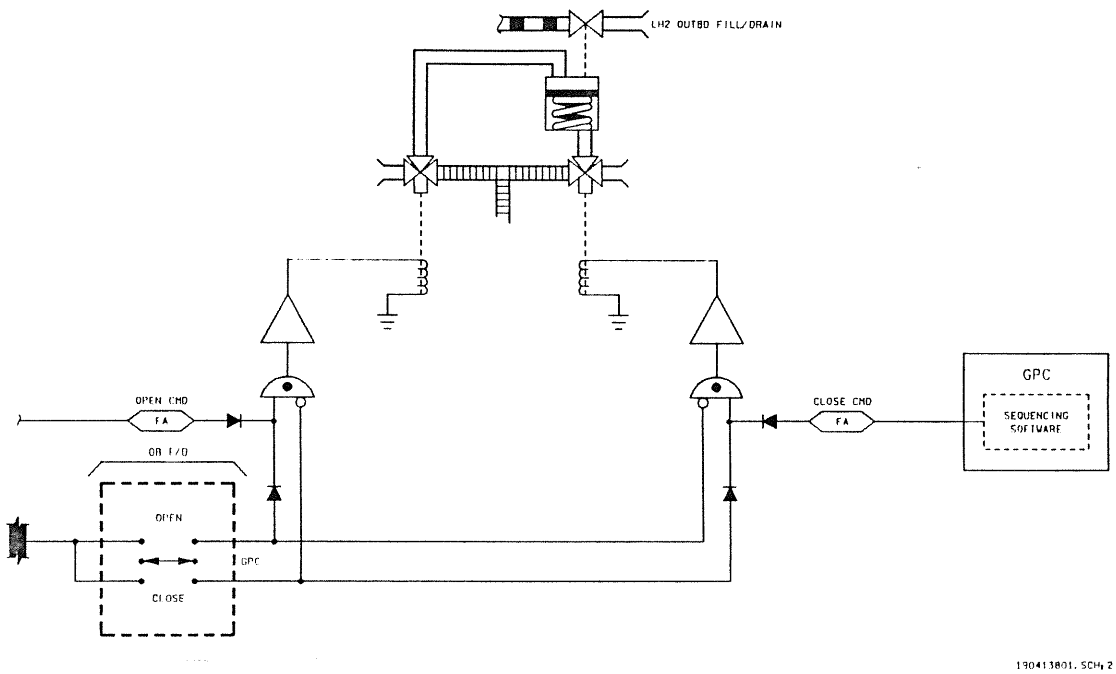


Figure 6.2-1.- Control for a typical MPS valve.

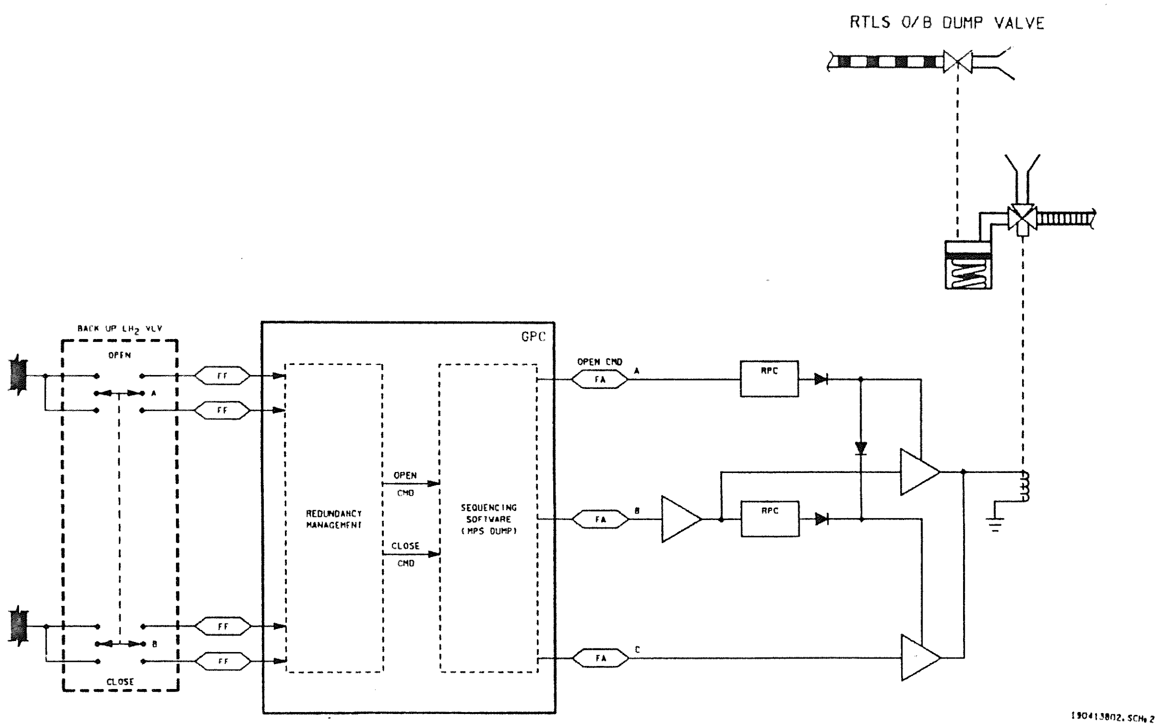


Figure 6.2-2.- RTLS outboard dump valve.

TABLE 6.2-I.- SW RM SELECTION FILTERS

Contact available:

Three

A B C	Output
0 0 0	0
0 0 1	0
0 1 0	0
0 1 1	1
1 0 0	0
1 0 1	1
1 0 1	1
1 1 1	1

Majority vote
two-of-three

Two

A B	Output
0 0	0
0 1	0
1 0	0
1 1	1

"logical and"

One

A	Output
0	0
1	1

one-of-one

6.2.3 Dilemmas

Failures in the switch or contact power supplies can cause dilemmas in the voting logic to occur. There are two types of dilemmas: (1) more than one switch position is indicated, and (2) no switch position is indicated. It would take at least two failures to occur to cause a dilemma for a two- or three-contact switch. It would take only one failure to occur to cause a dilemma for a one-contact switch. When a dilemma occurs, a default switch position is sent from the redundancy management to the sequencing software. For certain switches, the PASS will set a fault message that will be displayed to the crew to indicate that a dilemma has occurred and that the default switch position has been selected. See table 6.2-II for example dilemmas on a hypothetical two-position, three-contact switch.

TABLE 6.2-II.- SWITCH DILEMMA EXAMPLES

Contacts Available: Three | Two | One

(1) More than one position is indicated

Three		Two		One	
MAN	1 0 1	MAN	1 X 1	MAN	X X 1
AUTO	1 1 1	AUTO	1 X 1	AUTO	X X 1

(2) No position is indicated

Three		Two		One	
MAN	0 0 0	MAN	0 X 0	MAN	X X 0
AUTO	1 0 0	AUTO	1 X 0	AUTO	X X 0

Note: X indicates commfaulted MDM.

6.2.4 MPS Switches

Tables 6.2-III and 6.2-IV list the multicontact switches in the MPS subsystem that interface with the redundancy management software. The tables include the power source for each contact, as well as the FF MDM that receives the status of the contact. The tables give the default switch position that will be selected if a dilemma occurs. Only some of the three-contact switches have default positions. The ET and SRB SEP pushbuttons do not have default switch positions. The ET SEPARATION MAN/AUTO switch defaults to manual for OPS 1 and defaults to auto in OPS 6 for the RTLS case. The fault message that is displayed when the ET SEP switch has a dilemma is "ET SEP MAN" for OPS 1 and "ET SEP AUTO" for OPS 6. If the dilemma occurs in OPS 1, the message will appear. IF RTLS is subsequently selected, the "ET SEP AUTO" message will also appear. If the switch dilemma occurs on a nominal ascent and the default manual mode is selected, it can be overridden by a keyboard entry on SPEC 51. ET SEP AUTO SELECT is item 38 on the PASS display SPEC 51 and is item 28 on the BFS display SPEC 51. Only the PASS will detect a dilemma, select the default, and annunciate the fault message. If the default is set to MAN and the BFS is subsequently engaged, the default can be overridden on the BFS SPEC 51.

The only other MPS switch that displays a fault message is the two-contact MAIN ENGINE SHUTDOWN pushbutton. The message is "ME SHDN SW C(L,R)." The shutdown pushbuttons each generate two outputs to the SSME OPS software sequence: (1) MAIN ENGINE SAFING COMMAND, and (2) MAIN ENGINE SHUTDOWN COMMAND. The shutdown command will only be true if the safing command is true and no commfaults exist on either of the two inputs for the pushbutton. This means that for only one failure, the shutdown command cannot be sent. The engine must be shut down via the controller ac power switches on panel R2. If a commfault does exist and the engine goes out behind a data path failure, the pushbutton is pushed to send the safing command to close the prevalves and to mode guidance into its engine-out routines. If a control bus fails and the shutdown pushbutton is needed, the appropriate FF MDM must be commfaulted to mode guidance. The tables show the appropriate MDM to commfault for any control bus failure. Note that commfaulting an FF MDM will also drop the IMU on that string.

If control bus CNTL AB3 or CNTL BC3 fails, the MPS Propellant Dump Sequence switch fails. Instead of commfaulting the appropriate FF MDM to recover the switch, a dummy OMS burn is performed to start the dump sequence. This procedure sets the OMS ignition flag (which starts the dump) without actually firing the OMS engines. Note: The dummy OMS burn procedure will not work if BFS is engaged.

TABLE 6.2-III.- THREE-CONTACT SWITCHES

No. of MDM's commfaulted	No. of contacts available	Selection filter
0	3	Majority vote (two-of-three)
1	2	Logical and
2	1	Status of contest (one-of-one)

Redundancy level = 3 three-contact switches	position - contact		Power source		MDM	DEFAULT*
Main engine limit SD	Enable	- A	CNTL	BC2	FF2	
		- B		CA3	FF3	
		- C		BC3	FF4	
C3	Auto	- A		BC2	FF2	AUTO
S11		- B		CA3	FF3	
		- C		BC3	FF4	
	Inhibit	- A		BC2	FF2	
		- B		CA3	FF3	
		- C		BC3	FF4	
SRB separation	Man/Auto	- A	DSC	OF1	FF1	
		- B		OF2	FF4	MAN/AUTO
		- C		OF3	FF3	
C3	Auto	- A		OF1	FF1	
S1		- B		OF2	FF4	
		- C		OF3	FF3	
C3	SEP	- A	DSC	OF1	FF1	
S2		- B		OF2	FF4	----------
		- C		OF3	FF3	
ET separation	Man	- A	DSC	OF1	FF1	NOTE
		- B		OF2	FF4	MAN A
		- C		OF3	FF3	
C3	Auto	- A		OF1	FF1	(AUTO for RTLS)
S3		- B		OF2	FF4	
		- C		OF3	FF3	**
C3	SEP	- A		OF1	FF1	
S4		- B		OF2	FF4	----------
		- C		OF3	FF3	

* DEFAULT is used when: (1) more than one position is indicated
(2) no position is indicated

** Fault messages: ET SEP MAN (OPS 1)
ET SEP AUTO (OPS 6)

TABLE 6.2-IV.- TWO-CONTACT SWITCHES

No. of MDM's commfaulted	No. of contacts available	Selection filter
0	2	Logical and
1	1	Status of contact (one-of-one)

Redundancy level = 2 two-contact switches	position - contact		Power source		MDM	NOTE
Main engine shutdown* C3 S13	CTR	- A	CNTL	AB1	FF1	B
		- B		AB2	FF2	
S12	LEFT	- A		BC2	FF2	B
		- B		CA2	FF3	
S14	RIGHT	- A		CA3	FF3	B
		- B		BC3	FF4	
PRPLT DUMP SEQ	START	- A		AB3	FF1	C
		- B		BC3	FF2	
R2 S1						
	STOP	- A		AB3	FF1	
		- B		BC3	FF2	
BACKUP LH_2 VLV	OPEN	- A		AB2	FF3	
		- B		BC2	FF4	
R2 S2						
	CLOSE	- A		AB2	FF3	
		- B		BC2	FF4	

* Fault message = ME SHDN SW C(L,R)

NOTES

A. If the default occurs during nominal ascent, the default position can be overridden by a keyboard entry (ET SEP AUTO SELECT).

SPEC 51	OVERRIDE	
	PASS	BFS
	Item 38 Execute	Item 28 Execute

B. The inputs from the main engine shutdown pushbuttons will generate two outputs to the SSME OPS software sequence per engine:

(1) Main engine safing command

(2) Main engine shutdown command

The main engine shutdown command output will be true only if the main engine safing command is true, and no commfaults exist on either of the two-contact inputs of the shutdown pushbutton.

C. For control bus failure on CNTL AB3 or CNTL BC3 and no OMS 1 burn required (direct insertion) or a delayed OMS 1 burn is required, perform the MPS dummy OMS burn and the OMS ignition flag will start the MPS dump software sequence.

D. Fault messages for switch dilemmas occur in PASS only.

6.2.5 BFS

This section describes the redundancy management differences between BFS and PASS. In the software requirements, there is no separate redundancy management function in the BFS. Instead, each software sequence performs voting logic on multicontact switches. There is no commfaulting in the BFS, so the selection filtering will not downmode from two-of-three to "logical and" to one-of-one as the PASS. Not all of the MPS switches are monitored by the BFS. Table 6.2-V shows the ones that are monitored and the voting logic for each.

The contacts for the Main Engine Limit switch are voted two-of-three. If a dilemma occurs such that more than one position is indicated, the position with the highest priority is selected. The AUTO position is priority one, the ENA position is priority two, and the INH position is priority three. If a dilemma occurs such that no position is indicated, the AUTO position is assumed.

Since the SRB SEP MAN/AUTO position is not monitored by the BFS, no dilemma is possible. The SRB SEP AUTO contacts A, B, and C are voted two-of-three in the following manner:

If (A & B) or (A & C) or (B & C) are false, then the MAN/AUTO position is selected, otherwise the AUTO position is assumed.

Since the ET SEP AUTO position is not monitored by the BFS, no dilemma is possible. If the PASS detects a dilemma, it will default the switch to MAN and display the "ET SEP MAN" fault message. If the BFS is subsequently engaged, the manual position can be overridden by item 28 on SPEC 51.

Table 6.2-VI summarizes the differences between BFS and PASS.

TABLE 6.2-V.- BFS REDUNDANCY MANAGEMENT

Switch	Position	Sequence	Voting
Main engine limit shutdown (note 1)	ENABLE A,B,C	SSME OPS	(A&B) or (A&C) or (B&C)
	AUTO A,B,C	SSME OPS	(A&B) or (A&C) or (B&C)
	INHIBIT A,B,C	SSME OPS	(A&B) or (A&C) or (B&C)
SRB separation (note 2)	AUTO A,B,C	SRB SEP	(A&B) or (A&C) or (B&C)
	SEP A,B,C	SRB SEP	(A&B) or (A&C) or (B&C)
ET separation (note 3)	MAN A,B,C	ET SEP SEQ	(A&B) or (A&C) or (B&C)
	SEP A,B,C	ET SEP scheduler	(A&B) or (A&C) or (B&C)
Main engine shutdown	CTR A,B	SSME OPS	Both A & B
	LEFT A,B	SSME OPS	Both A & B
	RIGHT A,B	SSME OPS	Both A & B
PRPLT dump sequence	START A,B	MPS DUMP	Both A & B

Note 1: Dilemmas - More than one position indicated
choose AUTO over ENABLE
choose ENABLE over INHIBIT

- No position indicated
choose AUTO

Note 2: If (A&B) or (A&C) or (B&C) = FALSE, then MAN/AUTO = TRUE, else, AUTO = TRUE

Note 3: If PASS sets "ET SEP MAN" DEFAULT
BFS SPEC 51, ITEM 28 will override back to auto

TABLE 6.2-VI.- RM CAPABILITIES LOST AT BFS ENGAGE

1. No commfaults - no selection filter downmode
2. No dummy OMS burn procedure
3. No default switch position for dilemma
4. No fault messages for dilemma
5. BACKUP LH_2 VLV switch is not monitored - not functional
6. SRB SEP - MAN/AUTO switch position is not monitored
7. ET SEP - AUTO switch position is not monitored

6.3 I-LOAD ARMING MASS AND ONBOARD VEHICLE MASS CALCULATION

6.3.1 Background

There are eight low-level sensors that monitor the level of LO_2 and LH_2 in the external tank (ET). Four of the sensors are located at the bottom of the ET and provide warning against low LH_2 levels. The remaining four are in the 17-inch LO_2 feedline. The LO_2 sensors were once located in the ET. After STS-12, it was found that the amount of LO_2 remaining in the system at MECO was less than previously thought. The sensors were then moved to their present location.

The GPC's send a shutdown command to the SSME's when the low-level sensors indicate that LO_2 or LH_2 are near depletion.

To prevent a premature SSME shutdown due to failed dry sensors, the low-level sensors are not armed until an arm command is received from guidance.

The ET low-level sensor arm command is initiated by the ascent second-stage guidance function or the powered RTLS guidance function. From either source, the arm command is initiated under the following conditions:

1. Two SSME's have failed

2. Current vehicle mass is less than I-loaded mass

   * M < MASS_LOW_LEVEL (V97U4432C)

The arming mass I-load of 32,000 lb ensures the shuttle has the capability to at least make an AOA in the event of MECO at the arming mass. The arming mass is used to protect against GN&C dispersions, MPS loading errors, mixture ratio dispersions, and MPS system failures. The arming mass (MASS_LOW_LEVEL) is equivalent to (nominal zero weight margin - nominal flight performance reserve + MPS protection + estimated mass track error)/32.174. The current MPS and mass track error protection requirements for a two-engine press to MECO are:

Mass bias for 3-sigma performance dispersion	5,600 lb
Allowance for 1 guidance cycle (1.92 sec)	3,600 lb
Allowance for signal path delay	75 lb
Weight growth	2,000 lb
Mass tracking error	1,600 lb
MPS system failures	12,475 lb
Total	25,350 lb

An additional 6650 lb was added to the total to eliminate I-load changes because of differences in mission trajectories.

Failures in the MPS system require the most mass bias at 12,475 lb, or 40 percent of the I-load total. The MPS bias protects against the following failures:

	Mass lost, in lb
LO2 feedline relief system	6200
LH2 feedline	640
LH2 RTLS dump system	1800
LH2 Hi-point bleed system	285
LO2 overboard bleed system	3000
LH2 flow control valve system	550

The nominal zero margin weight (NZMW) is equivalent to:

NZMW = Weight of orbiter without consumables plus the weight of:

+ nonpropulsive consumables at SRB ignition
+ left OMS fuel
+ right OMS fuel
+ left OMS oxidizer
+ right OMS oxidizer
+ forward RCS
+ aft RCS
+ cargo module
+ ET
+ ET bouyancy
+ MPS pressurant
+ ice, frost, and liquified air
- estimated shuttle system losses at MECO
+ unusable MPS propellant
+ unusable MPS propellant reserves
+ MPS propellant vented after SSME valve closure

Note: All but the last three weights in the equation above are obtained from each flight's Vehicle Summary Weight Statement in the Trajectory Design Data Pack (TDDP). The last three weights are contained in the nominal MPS propellant inventory in the TDDP.

6.3.2 Guidance Calculation of Current Vehicle Mass

First Stage

From liftoff to SRB separation, the vehicle mass calculation is a first-stage guidance input task. Expelled mass (SSME only) is subtracted from an I-loaded liftoff weight, as follows:

DTGD = T_NAV - T_NAV_PREV

T_NAV_PREV = T_NAV

(1) M = M - 0.01 * (DTGD * K_CMD * MDOT_SSME * N_SSME)

Where:

T_NAV and T__NAV_PREV are current and previous timetags indicating the the time difference since the last mass update.

KCMD = Commanded thrust force in percent

N_SSME = 3 - number of engine-out flags set

MDOT_SSME (V97U4442C) = I-loaded value of 32.22215576 slugs/sec

Equation 1 indicates that the first-stage mass track calculation (three engines on) is a function of commanded power level only. If actual engine power levels are different than commanded, the mass track will be in error.

After SRB separation, the current vehicle mass (as calculated by first-stage guidance) is used to initialize the second-stage mass update task.

Second Stage

The mass update procedure is a general purpose guidance task. A current vehicle mass is estimated by the second stage guidance function and compared to an I-load vehicle mass. Mass estimation is accomplished by an exponential extrapolation (approximated by the first two terms of the Taylor series expansion) of sensed velocity divided by equivalent exhaust velocity:

M_{new}=M*[2.0/{1.0+0.5*DVSMAG/VEX}-1.0]

M_{new} is current mass estimate

M is last mass estimate

DVSMAG = ABVAL (VS - VSP) => Delta sense velocity
VS = Current value of accumulated sensed velocity
VSP = Previous value of accumulated sensed velocity

(2) VEX is equivalent exhaust velocity

If the current estimation of mass does not vary significantly from the past estimate the mass value is not updated:

- Update mass if (M - M_{new}> MUP_TH = 10 lbm)

6.3.3 Guidance Routines Used To Determine Vehicle Mass

The mass update task calls the thrust parameters task (THRST PRM TSK) for estimates of thrust force, mass flow rate, and equivalent exhaust velocity:

(3) $FT = \underline{.01\ KCMD\ N\_SSME\ FT\_SSME} + (N\_OMS\ F\_OMS + N\_RCS\ FT\_RCS)$

(4) $MDOT = \underline{.01\ KCMD\ N\_SSME\ MDOT\_SSME} + (N\_OMS\ MDOT\_OMS + (N\_RCS+N\_RCS+N\_RCS\_NULL)\ MDOT\_RCS$

(2) VEX = FT/MDOT

KCMD = Commanded thrust force in percent

N_SSME = 3 - number of engine-out flags set

FT_SSME (V97U4392C) = Nominal I-loaded SSME thrust level (470,000.0 lb)

MDOT_SSME (V97U4442C) = I-loaded value of 32.22215576 slugs/sec

FT = Thrust force

MDOT = Mass flow rate

Since no OMS or RCS propellant is used during first and second stage, only the underlined portions of equations 3 and 4 are important here. The output of equation 2, which is a THRST PRM TSK, is used by the second-stage guidance mass update task.

6.3.4 Crew Displays

The crew does not have insight into the vehicle mass or low-level arm command. The percent of total propellant remaining is displayed on the crew pass ascent trajectory display.

The percent of propellant remaining is calculated using the following:

CURRENT_PRPLT (V97U6913C) = 32.174 (M - R2_FINAL_WT)

PRPLT (V97U6927C) = (CURRENT_PRPLT/TOTAL_PRPLT) * 100%

R2_FINAL_WT = 312,886 lb

TOTAL_PRPLT = 1,593,603 lb

R2_FINAL_WT and TOTAL_PRPLT vary from flight to flight.

6.3.5 Mass Track Errors

This vehicle mass calculation and propellanr remaining computation can compensate for one or two engines out; however, they were not designed to compensate for engine performance anomalies. Furthermore, the computations are optimized for the RTLS case; i.e., the OMS propellant is dumped (R2_FINAL_WT = R2_FINAL_WT for RTLS). As a consequence, engine performance cases affecting the overboard mixture ratio will introduce error into the calculation.

Analysis of worst case scenarios, specifically two engines out at single-engine press to MECO and the third engine low by 10 sec in ISP, indicates that the onboard propellant computation is never off by more than 1.5 percent. For this case, when the onboard computation indicates 2 percent, the actual propellant remaining is 0.5 percent. Therefore, for any case where manual engine throttling is to be performed at 2 percent, throttling will occur before propellant low-level cutoff. The LO_2 low-level sensors indicate less then 0.2 percent propellant remaining.

6.3.6 References

1. Diane Walyus and Gerry Gottselig, "Low Level Arming Mass Determination," March 27, 1987

2. Booster Systems Briefs, Basic, Rev A

3. STS83-0002C, Functional Subsystem Software Requirements

4. SODB, JSC-08934, Rev E, vol 1

6.4 LO_2 GEYSER PREVENTION DURING LOADING

6.4.1 Background

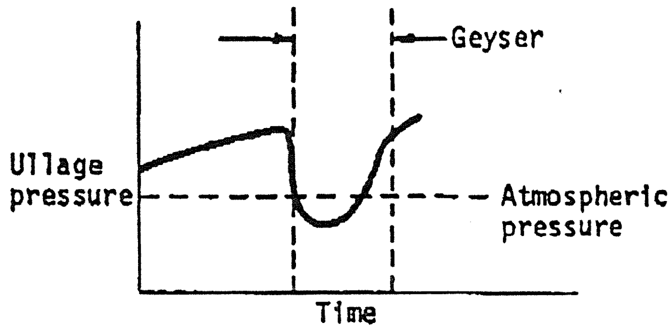
Between August 1976 and June 1977, a full-scale simulation of the space shuttle antigeyser system was tested at the Engineering Propulsion Laboratory, Martin Marietta Corporation, Denver, Colorado. The function of the space shuttle antigeyser system was to preclude the formation of geysers in the main propulsion system LO_2 feed system. The LO_2 feed system on the space shuttle consisted of an orbiter manifold and feedline, an ET main feedline, and an ET antigeyser line. The baseline space shuttle antigeyser system consisted of the ET main feedline and the ET antigeyser line.

6.4.1.1 Geyser Phenomenon

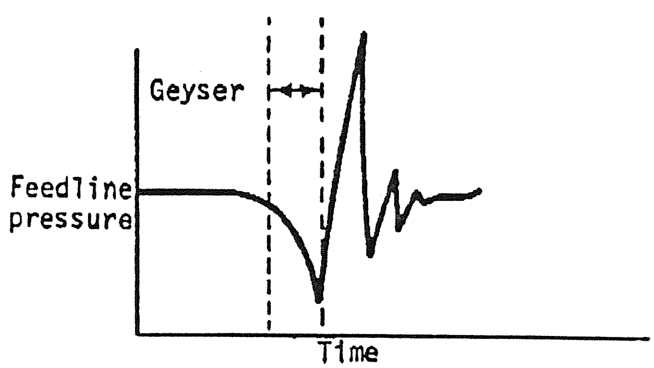
The geysering phenomenon is the rapid expulsion of a boiling liquid and its vapor from a vertical tube. For example, in the case of a long vertical column of fluid connected to a reservoir, such as the suction line and the LOX tank for a launch vehicle, the line is subject to ambient condition heating from top to bottom. As this heating begins, the fluid adjacent to the wall is warmed and becomes less dense than the fluid in the center of the line. Because of this density difference, a convection pattern is established, and warm fluid rises along the tube wall while cooler fluid from the reservoir descends down the center of the line to keep the system in equilibrium. As heating continues, a boundary layer is established along the wall that grows in thickness with time. This layer also grows in thickness from the bottom of the tube toward the top of the tube. If the heating rate is sufficient to cause the boundary layer to grow and fill the tube, the cool fluid flow from the reservoir is stopped, thereby halting the convention pattern. With the cessation of convection, additional heating causes the temperature to rise in a fixed amount of fluid. Eventually, this fluid becomes saturated and general boiling starts bubble generation. The bubbles form on the wall and then detach and start to rise in the liquid. As they rise, they coalesce and form a large "Taylor" bubble. The formation of the large bubble results in reduced pressure below the bubble and, consequently, more bubbles are formed in the saturated liquid. This self-sustaining reaction occurs rapidly and forms vapor faster than it can escape the tube. As a result, the vapor is rapidly and violently expelled from the tube in the form of a geyser. The geyser and subsequent water hammer effect, caused by rapid tube refill from the reservoir, can cause damage to the system.

6.4.1.2 Geyser Effect

The explosion of cryogenic liquid from the feedline into the vehicle tank by the rapidly rising bubble, caused by propellant heating, generally does not result in damage to the tank or feedline. However, if this explosion occurs when the tank contains a large warm ullage, implosion of the propellant tank can occur. This results because of the rapid cooling and contraction of the warm ullage gas, caused by large quantities of cold liquid droplets being expelled into the ullage by the geyser. This rapid cooling and contraction causes the tank pressure to fall below atmospheric pressure as shown in the sketch below, at which time tank implosion is possible. This condition is prevented by the installation of a simple deflector over the tank outlet to deflect the expelled liquid and prevent rapid ullage gas cooling. Once the tank is filled with liquid, and the ullage is small and cold, this geyser effect presents no problem.



The major geysering effect that may cause vehicle damage does not occur during the geyser, but during feedline refill immediately following the geyser. When the geyser occurs, the feedline is emptied of essentially everything except cold propellant vapors. As the line refills with cold liquid, the vapors condense at the interface of the liquid. Since the vapors condense, they provide no cushioning for the falling liquid, such would be the case of water filling a tube containing air. The impact pressures at the bottom of the line, resulting from the uncushioned refill can be extremely high and are unpredictable. This water hammer effect is shown in the sketch below. The high pressure resulting from the



Effect of line refill on feedline pressures refill can result in catastrophic failure of low pressure, lightweight, airborne feedline. Since geysering results in conditions that may severely damage the vehicle tankage or propellant-feed system, it must be prevented for all conditions.

6.4.1.3 Shuttle Geyser Suppression

To prevent geysering, the liquid in the feedline must be maintained in a subcooled condition. This can be accomplished by several methods. In a program under contract NAS10-7258, Propellants and Gases Handling in Support of Space Shuttle, results of several geyser suppression techniques were evaluated. A summary of Rockwell concepts evaluated under contract NAS10-7258, and major comments, are included in paragraph 6.4.12 at the end of this section. This study used an experimentally determined correlation curve to predict the geyser potential of candidate space shuttle feedline designs. The results of this evaluation indicated that potential geysering problems existed only in the LO_2 feedlines. The results of this evaluation also indicated that propellant recirculation by a concentric line was the competitive candidate. Subsequent to the analysis, a prototype vertical concentric feedline was fabricated and tested. It consisted of a 40-foot vertical feedline composed of a 5-inch-diameter inner line (core), and an 8-inch-diameter outer line with an 850-gallon propellant tank. LN_2 and LO_2 were used as the test fluid. The test proved that this concept would prevent geysering in a vertical line and would provide cold liquid at the bottom of the line by self-circulation down the core and up the annulus.

6.4.2 Baseline Antigeysering System

The baseline antigeyser system is designed to maintain low temperatures in the LO_2 main feedline by continuously circulating liquid drawn from the propellant tank down the main feedline, up a smaller line, and back to the tank. This is accomplished by injecting helium into the lower end of a 4-inch-diameter antigeyser line. The rising helium bubbles in the anti-geyser line induce a bulk liquid flow up the line, which is replaced by liquid drawn from the feedline.

Several distinct flow modes can be described for the baseline antigeyser system. In the normal flow mode, the liquid flow in the main feedline is directed from the LO_2 tank downward toward the aft end of the main feedline. This condition prevails as long as the LO_2 flowrate from the orbiter (ORB) to the ET through the main feedline disconnect is less than the flowrate induced up the antigeyser line by injected helium. When the flowrate from the ORB to the ET exceeds the liquid flowrate induced up the antigeyser line, the excess is required to flow up the main feedline from the aft end toward the LO_2 tank. Operation of the antigeyser system in this manner is referred to as split flow.

When the flowrate from the orbiter to the ET is exactly equal to the liquid flowrate up the antigeyser line, the liquid flowrate in the main feedline is zero, and it is said to be stagnated. Under these conditions, the liquid in the main feedline warms under the influence of the heat conducted through the feedline wall, local saturation may occur and geysering can result.

6.4.2.1 Main Feedline Injection

An alternative LO_2 antigeyser system for the space shuttle is main feedline injection. It does not employ an antigeyser line to provide a flow loop, but consists instead of the injection of helium directly into the aft portion of the main feedline. Local saturation is prevented by the evaporation of LO_2 into the rising helium bubbles. As the bubbles of helium rise, their volume is increased as the local pressure in the line is decreased. Evaporation of LO_2 into the bubble continues, keeping the bubble saturated with LO_2 vapor. Unlike the baseline system, the main feedline injection antigeyser system does not have flow regimes. Its effectiveness depends on the flowrate of helium injected into the main feedline and on the flowrate of liquid in the main feedline. An additional variable of importance is the local saturation pressure in the main feedline. The bubble volume at any location is inversely proportional to the difference between the local pressure at that location and the saturation pressure corresponding to the temperature at that location. Thus, the more closely the fluid approaches local saturation and the more rapidly the bubble size increases, the more the increased LO_2 vaporization rate makes the evaporative cooling process effective in preventing such saturation.

Although concentric line recirculation was initially thought to be the most effective system to prevent geysering in the shuttle feedline system, further evaluation of antigeysering systems under NASA contract NAS10-7258 showed the following:

- The main feedline injection configuration was capable of indefinite hold with no liquid flow from ORB to ET, provided helium injection was maintained. In the absence of helium injection, main feedline temperatures were observed to increase at a rate of 4° R/hr, resulting in a hold capability of 3 to 4 hours.

- The main feedline injection configuration provided safe operation during reverts, reinitiation from either the inboard or outboard fill-and-drain valves, and helium bubble injection.

6.4.3 Refining the Space Shuttle Helium Injection Antigeysering System

The baseline geyser configuration resulted from a worst case geyser experienced during the antigeyser test program. This geyser was intentionally caused by employing the minimum helium injection flowrate, operating with the tank vented, and introducing 178° R LO_2 into the main feedline at a flowrate slightly in excess of the flowrate induced up the antigeyser line. Thus, warm propellant flowed slowly up the main feedline toward the LO_2 tank.

The magnitude of the refill pressure spike was inadequately measured by the transducer. The magnitude measured with a high-response transducer was 524 psia (table 6.4-I). Based on the time between the passage of the first warm fluid particle at the ORB/ET interface, the time of geyser occurrence, and the flowrate of the fluid up the feedline, it was estimated that this warm fluid particle was between the middle and top of the feedline vertical section at the time of the geyser. The subcooling plots of the data taken at the middle and top of the feedline vertical sections indicated that the warm fluid particle had passed the middle of the vertical section but had not yet reached the top at the time of the geyser, thus confirming this estimate.

A main feedline injection configuration geyser also occurred during a filling operation after the system had been shut down to repair a ruptured, burst disc. Thus, the facility was warmer than cryogenic temperatures but, overall, had not warmed to ambient temperatures. The helium injection flowrate was 0.011 lbm/sec and the ORB to ET flowrate was 20 lbm/sec. As soon as the feedline filled, pressure surging began. This surging decayed and stopped momentarily and then began again, indicating a growing magnitude with time. Examination of the incoming fluid temperature indicated that the interface temperature began at 180° R, decayed to 173° R over 4 minutes and then increased again to 176° R, after which it remained relatively constant. Analysis of fluid particle displacement over time indicated that geysering ceased as the 173° R fluid particle passed into the horizontal portion of the feedline prior to exiting into the LO_2 tank.

A significant (approximately 10° R) reduction of the liquid temperatures took place as it rose in the feedline because of evaporation of LO_2 into helium bubbles rising in the feedline. Geysering continued as the warmer fluid entered this horizontal section of the feedline. Geysering ceased when the fill operation was stopped and cold fluid was drawn out of the LO_2 tank into the feedline under the action of the engine bleed flowrate.

Analysis of the feedline temperatures indicate that the effect of nitrogen dissolved into the LO_2 in the feedline resulted in a 3° to 5° R reduction of the local fluid saturation temperature.

TABLE 6.4-I.- ULLAGE PRESSURES

Event	Location	Low response transducer (psia)	High response transducer (psia)
Baseline	MFL bottom	81	423
LO_2 geyser	MFL middle	41	---
1/26/77	AGL bottom	81	423
(See fig. 6.4-2)	Orbiter simulator	120	524
MFL ins.	MFL bottom	70	---
LO_2 geyser	MFL middle	56	---
6/02/77	AGL bottom	70	---
(See fig. 6.4-7)	Orbiter simulator	78	---
	Pump discharge	70	262
Baseline	MFL bottom	53	408
LN_2 geyser	MFL middle	21	360
11/04/76	AGL bottom	53	398
(See fig. 6.4-10)	Orbiter simulator	56	453

6.4.4 Temperature Limits During Childown

The childown requirements were totally revised following the mini-geyser that occurred during the fill to 2 percent during main propulsion system test article (MPTA) SF12, the first test without an antigeyser line. A new procedure was developed to close the inboard fill and drain valve (VP10) for 5 minutes after liquid reached the early cutoff (ECO) sensor at station 1616, while flow was continued through the tail service mast (TSM) vent valve.

Once temperature limits for fill to 2 percent were established, based on the SF12 data, the same limits were applied to the childown period. One exception was the middle feedline temperature, which had no limit because the temperature probe was not fully covered with liquid during childown. Another exception was the orbiter inlet temperature, which had a -291° F limit during fill to 2 percent and a -288° F limit during childown. The warmer limit for the orbiter inlet temperature during childown was chosen, based on results of KSC test data that showed temperatures up to -288° F at the start of childown.

This procedure was used successfully for MPTA T1-04 (without engines installed) and resulted in peak feedline temperatures that were 3° F colder than SF12. This procedure was repeated for STS-4, the last ET with the antigeyser line on. Test results showed that with the engines installed, the 5 minute PV10 closure time would not be sufficient; the time was increased to 10 minutes for STS-5. In anticipation of the future change in location of the ECO sensors, the use of the ECO sensors as the signal to close PV10 was replaced with an equivalent orbiter inlet pressure of 22 psig or a skid outlet pressure of 55 psig. The latter was purposely biased 2 psig high to ensure that the orbiter inlet pressure reached its limit first, thus controlling operations.

Although this procedure worked fine for STS-5, it was not successful for the STS-6 FRF. When PV10 was opened, the MPS disconnect temperature redline limit was exceeded, and two stop flows were subsequently encountered. The cause was attributed to MLP-2, which had different replenish valve and TSM vent flow characteristics. These produced higher pressures while PV10 was closed, exceeding the cracking pressure across the valve and allowing the LO_2 in the uninsulated orbiter fill and drain line to remain liquid and warmed up. When PV10 was opened, the warm LO_2 was expelled into the orbiter manifold, exceeding the MPS disconnect temperature limit and resulting in a stop flow. During the stop flow, the TSM vent was not opened and the warm LO_2 remained in the orbiter because no liquid was flowing. When flow was reinitiated, redline temperatures were exceeded on the MPS disconnect and feedline temperatures, resulting in a second stop flow. Ten seconds later, a mini-geyser, which voided 34 to 40 ft of feedline, occurred.

To prevent a reccurrence during the second STS-6 FRF, the OMRSD was revised to change the nominal replenish valve position during PV10 closed period from 12 percent to a value that would produce an orbiter inlet pressure of no more than 35 psig, (similar to STS-5) and a flowrate no less than 75 gpm. This requirement was successfully verified during the FRF with a replenish valve position of 8 percent for MLP-2. A second OMRSD change was made in case temperature limits were exceeded during fill to 2 percent and a stop flow occurred. The new requirement was to open the TSM vent for a minimum of 4 minutes prior to reinitiation, thereby draining the hot slug from the orbiter.

Although no temperature exceedances were experienced during the second STS-6 FRF, temperature headroom was lower than STS-5. The cause was attributed to a higher orbiter heat leak on the new OV-99 (Challenger). The additional heat leak was estimated to be 1.8 Btu/sec (1 Btu/sec due to change in purge gas from N2 to H3, 0.8 Btu/sec due to removal of insulation from the orbiter half of the ET/orbiter disconnect). Consequently, the working group decided to revise the chilldown procedure to acquire additional headroom during the fill to 2 percent. The new requirement lowered the orbiter inlet pressure level for closing PV10 from 22 to 15 psig. A psi reduction in pressure corresponded to a 3° F reduction in saturation temperature. This modification was successfully tested during the STS-6 launch countdown. The temperature headroom was increased by approximately 1° F.

6.4.5 Temperature Limits During Fill to 2 percent

Temperature limits for the fill to 2 percent phase were based on peak temperatures obtained during MPTA SF12, as shown in table 6.4-I. Since a mini-geyser occurred during the SF12 fill to 2 percent, the OMRSD limits were set 1° F colder than SF12 to ensure that any future mini-geyser would result in lower press surge. The OMRSD limits were further lowered by an additional 1° F to account for a possible error in the MPS instrumentation. The resulting limit of -288.5° F at the middle feedline temperature was conservatively rounded off to the lower value of -289° F to accommodate the inability of the KSC display to indicate tenths of a degree.

For LWT, the bottom feedline temperature sensor was moved from station 1965 to 2044. The OMRSD limit for this new temperature location for LWT was determined from MPTA test T1-04, where both temperature probe locations were instrumented. A comparison of the T1-04 test data showed that the temperature at the new (LWT) location was 2° F higher than the old HWT locations during chilldown and the first half of fill to 2 percent (including the peak temperature period). Consequently, the OMRSD limit for LWT was increased 2° F from the HWT temperature limit for both chilldown and fill to 2 percent loading phases.

As discussed previously, chilldown procedures were revised several times in an attempt to reduce the peak hot slug temperatures that developed in the orbiter during chilldown and traveled up the feedline during fill to 2 percent. These peak temperatures were less than 1° F colder than OMRSD limits. This resulted in bumpy boiling after the hot slug passed the mid feedline temperature sensor and reached the horizontal section of feedline, where saturation temperatures were exceeded and feedline pressures showed fluctuations of ±2 to 6 psi, depending on peak hot slug temperature. An example of exceeding the OMRSD limits occurred during the first STS-6 FRF when a hot slug was observed to travel up the feedline. Shortly after the mid feedline temperature reached a peak value of -287° F (0.5° F colder than SF12 but 2° F warmer than the OMRSD limit) a mini-geyser similar to SF12 was observed as feedline pressure dropped about 17 psi and voided 35 to 40 ft of feedline. No refill water hammer resulted because there was no liquid in the tank.

A simplified LO_2 MPS chilldown procedure, expected to produce additional headroom (temperature margin) during fill to 2 percent, was developed by the working group and tested during STS-7. The new procedural steps were as follows:

During MPS chilldown, LO_2 was loaded at approximately 300 gpm with the TSM vent open. When the orbiter inlet pressure reached 34 psig (approx. X_T 1300), the replenish valve was closed, allowing the hot LO_2 in the feedline and orbiter manifold to drain through the TSM vent line. When the orbiter inlet pressure reached 10 psig, the replenish valve was opened and the TSM vent was closed, initiating fill to 2 percent.

During fill to 2 percent, each SSME inlet temperature was required to be colder than -280° F (preburner discharge temperature 205° F as backup) when the orbiter inlet pressure was greater than 35 psig (skid outlet press 57 psig as backup). The purpose of this requirement was to ensure that bleed flow through each engine had begun before liquid entered the tank. This ensured that all engine bleeds were functional. There were several advantages to this procedure:

1. Leaving PV10 open allowed the hot LO_2 in the ET feedline and orbiter manifold to drain out the TSM vent line. Keeping PV10 open also avoided potential problems annunciated with closing it. The problems included high pressure in the fill and drain line that allowed LO_2 to warm up and possibly form a geyser through PV10 into the orbiter when cracking pressure across PV10 was exceeded.

2. Increased head pressure at 34 psig (instead of 18-22 psig) and assisted in establishing SSME bleed earlier.

3. Simplified loading procedure accommodated facility differences between KSC, NSTL, and VAFB.

6.4.6 Temperature Limits During Fast Fill

Temperature limits during fast fill are listed in table 6.4-II (OMRSD table S00FD.091). The temperature limits vary linearly between -291° F at 2 percent (with 2 psig ullage pressure) to -281° F at 100 percent. The temperature limits apply to sensors at the orbiter inlet, MPS disconnect, and ET feedline (bottom and middle). All read the same temperature at fast fill flow rates.

Since temperatures were close to the redline limit of -291° F for the transition to fast fill during STS-5, it was decided to increase head room by raising the limit to -290° F to accommodate a 1° F instrument error, while requiring additional ullage pressure to maintain the same saturation margin.

The critical region for the transition to fast fill is the upper vertical section of feedline near station 1030, where the feedline becomes near horizontal. Above that point, the LO_2 is cooled by convection and mixing with the cold liquid at the surface. Below that point, saturation temperature increases because of liquid head. Figure 6.4-11 shows the theoretical saturation line at 2 percent psig ullage. The warmest transition to fast fill occurred during MPTA T1-02, where a feedline temp sensor located at station 1097 recorded a peak temperature of 291° F at an ullage pressure of 2 psig. As indicated in figure 6.4-11, this temperature would have intersected the saturation line at station 1060, allowing boiling to occur above that point. Thus the original limit of -291° F at 2 psig ullage was set at the temperature/pressure combination that would allow boiling to occur above station 1060 (with no geyser verified by T1-02), while precluding boiling below station 1060.

TABLE 6.4-II.- LO_2 ANTIGEYSER TEMPERATURES °F, STS-9

	Fill to 2 percent[1]		Fill to 2 percent[2]		Fast fill start[3]		Topping		Replenish	
	Actual max.	Max. limit	Actual max.	Max. limit	Actual max.	Max. limit	Actual max.	Max. limit	Actual max.	Max. limit
Orbiter inlet	-294.3	-288	-294.3	-291[4]	-294.6	Note 5	-295.0	-281	-292.8	-281
Orbiter feed manifold	-292.8	-284	-291.0	-284	-293.0	Note 5	-293.6	-281	-293.4	-275
ET bottom	---	NA	-291.0	-285	-292.5	Note 5	-293.2	-281	-284.0	-276
ET middle	---	NA	-291.5	-289	-292.0	Note 5	-292.5	-281	-285.5	-280
Engine inlets	---	NA	-285.0	-280[4]	---	NA	---	NA	---	NA

Notes:

1. 10 to 22 psig at orbiter inlet plus TSM vent closed.
2. 22 psig at orbiter inlet to 2 percent level.
3. Actual ullage press 1.4 psig.
4. 35 psig at orbiter inlet to 2 percent level.
5. -291° F with ≤2 psig ullage; -290° F with ≥3 psig ullage.

However, no allowances had been made for instrumentation error, estimated to be as much as 1° F. To account for instrumentation error, several options were considered:

1. Lower the redline temperature limit by 1° F from -291° to -292° F.

2. Increase the ullage pressure requirement by 1.2 psi to raise the saturation temperature by 1° F.

3. Lower the point at which boiling is allowed to occur by about 3 ft from station 1060 to 1097. Option 2 was chosen by the working groups, since option 1 was not reliable and it was considered unlikely that a geyser would have a chance to develop in 3 ft.

Additional details of this analysis are given in table 6.4-I. It should be noted that the assumed density is conservative, since it does not consider the reduced bubble volume at fast fill flowrates.

The limit at 98 to 100 percent was conservatively chosen as -281° F to provide 3° F margin from saturation at station 1060 with Ø psig ullage, or 3° F margin from STS-47 station 1097 with Ø psig ullage and a 1° F allowance for instrumentation error.

6.4.7 Temperature Limits During Topping

Temperature limits during topping are the same as for the end of fast fill; that is, -281° F. This limit provides a 3° F margin from saturation in the critical area near the top vertical section of the ET feedline with the tank ullage pressure at 2 psig minimum to account for a possible 1° F instrumentation error.

6.4.8 Temperature Limits During Replenish

Three sets of limits are given in the OMRSD: engine start limits, operating limits, and saturation limits. The operating limits are the actual redline limits; the others are for reference only. Engine start and saturation limits are discussed below.

a. Engine Start Limits

These temperatures were the original OMRSD limits during replenish. They were also used as the initial temperatures for drainback predictions that established ICD limits for ET/orbiter interface and engine inlet temperatures during drainback and engine start.

The original temperature limits were based on "worst case" temperatures predicted by the math model simulation of main feedline helium injection, using the worst case MPTA orbiter heat leak of 15.8 Btu/sec and a maximum fill temperature of 169° F (-291° F). Net LO_2 replenish and helium inject flowrates were assumed to be nominal values of 1.2 and 0.01 lb/sec, respectively. The mid feedline temperature prediction was adjusted 2° F warmer to provide a margin for the predicted temperature to be as much as 2° F low in the middle of the feedline. Lower temperatures were observed in reconstructions of MPTA SF12 and T1-02 test data during replenish. This problem has not shown up on flight vehicles.

Reconstructions also showed that the maximum MPTA orbiter heat leak of 15.8 Btu/sec was too high. However, if the orbiter heat leak was reduced to the maximum SOFI heat leak of 13 Btu/sec and worst case replenish and helium inject flowrates (0.55 and 0.009 lb/sec, respectively) were used in the math model, the results were practically identical to the original worst case temperatures predictions.

b. Saturation Limits

KSC objected to using the engine start limits as redline limits during replenish because exceeding the limits would cause no immediate concern until time to start the engines. KSC held the position that the redline temperatures during replenish should reflect real limits, such as geyser-avoidance values.

Two sets of geyser avoidance limits were considered. One set provides a 3° F margin from saturation for a full tank with 2 psig ullage pressure. However, there were several concerns about using these relatively high temperatures as redline limits. The major concern was the potential of a single failure (replenish valve failed open) to push hot LO_2 rapidly up the feedline, where it would exceed saturation limits and result in a geyser before corrective action could be taken.

To preclude the possibility of a geyser because of a failed open replenish valve, the math model was used to determine the maximum allowed feedline temperatures that would maintain a 3° F margin from saturation limits when translated up the feedline. As previously discussed, the critical area with the least margin is the upper part of the feedline where the limit is -281° F, including a 3° F margin from saturation with zero ullage pressure. Assuming a replenish temperature of -281° F and a failed open replenish valve flowrate of 250 gpm, maximum allowed feed system temperatures predicted by the math model are as follows:

Middle feedline temperature (X1616) = -280° F

HWT bottom feedline temperature (x1965) = -277° F

LWT bottom feedline temperature (x2044) = -276° F

MPS disconnect temperature = -275° F

These geyser-avoidance limits were selected as the actual OMRSD redline temperature limits.

It should be noted that these temperature limits are above the -281° F requirement and depend on helium injection cooling and replenish flow rates of no greater than 250 gpm to reduce them to -281° F before reaching the upper part of the feedline.

6.4.9 Terminal Count Sequence (9-Minute Drainback)

OMRSD requirements for the terminal count sequence are as follows:

a. SOOFFO.141 (HWT) or SOOFFO.142 (LWT) - Verify flight mass okay before replenish terminate

b. SOOFMO.042 - Terminate replenish by closing replenish valve

- At T-535 sec, if 100 percent sensor used for replenish control
- At T-610 sec, if 100.51 percent sensor used for replenish control

c. SOOFMO.050 - Close PV10 immediately after LO_2 replenish

d. SOOFMO.060 - Open TSM vent/drain

e. SOOFMO.321 - Close ET vent valve, inhibit ogive stabilization

- Terminate helium injection within 50 sec of replenish termination

This sequence was recommended by KSC as the optimum sequence with minimum impact in case one of the vehicle or facility valves failed to respond. The termination of helium inject was purposely delayed until the last step. If it were turned off earlier in the sequence and a later valve failure occurred, there would be as much as a 5-minute delay after helium inject was reinitiated before it would reach the ullage volume.

The OMRSD requirement for the start of drainback (replenish termination) was originally the ICD time of T-540 seconds. This requirement was changed to T-535 seconds when it was discovered that, because of the programmed 10-minute hold at T-9 minutes, GLS might not be able to start the autosequence at exactly T-540 seconds. The termination of helium inject within 50 seconds of replenish termination was based on the estimated time to complete the intermediate steps. The actual requirement is the time required for the helium to clear the feedline to prevent excessive helium ingestion by the engines (4 percent by volume). The LCC requires helium to be off at T-310 seconds for a 9-minute drainback.

6.4.10 Terminal Count Safing and Recovery

If a launch is not performed (engine not started, engine abort, or FRF), there are two options, reestablish replenish or drain the external tank. In order to reestablish replenish, the tank must first be vented from prepress. Tank drain may be initiated with the vent open or closed. In order to vent the tank from prepress levels, helium inject must be turned on for at least 300 seconds before opening the vent valve, and prepress must be turned on concurrent with opening the vent valve. This allows sufficient time for helium to reach the ullage to prevent vent undershoot.

Antigeyser requirements after an engine shutdown for an on-pad abort are listed in S00E00.B90. The first step is to initiate helium inject within 15 sec. The sooner helium inject is reestablished, the sooner the tank can be vented. Next, orbiter bleed is reestablished within 2 minutes. Once bleed has been reestablished, the flow of cold LO_2 down the feedline through the engines prevents heat soakback from the engines and keeps the system safe from geysers. If orbiter bleed cannot be established, the math model predicts that heat soakback from the engines causes the engine inlets to heat up to saturation limits in about 12 minutes. Consequently, if bleed is not reestablished with 2 minutes, the OMRSD requirement is to isolate the SSME's by closing the prevalves and pogo return valves within 30 seconds to delay heat soakback to the orbiter from the SSME's. Tank drain is initiated within 12 minutes of the engine shutdown.

6.4.10.1 Stop Flow Reinitiation

Analysis of KSC stop flow data shows that a hot slug develops between the skid inlet and skid outlet. During the stop flow, the magnitude of the hot slug is not observable on skid inlet, skid outlet, or orbiter inlet temperature measurements. When flow is reinitiated, however, the hot slug is observed to pass first the skid outlet, then the orbiter inlet. The magnitude of the hot slug increases with time, and was observed to be 2.5° F for a 10-minute stop flow (STS-2), 4° F for a 16-minute stop flow (STS-1), and 7° F for a 33-minute stop flow (STS-4).

The temperature limitation for the hot slug is the same as the limits during fast fill. In fact, the same table (OMRSD table S00F00.091) is used. The reinitiation process requires that the temperature at the skid outlet be monitored continuously for 2 minutes at low flow (approximately 90 gpm). If the skid outlet temperature fails, the orbiter inlet temperature serves as a backup, but must be monitored continuously between 2 and 5 minutes at low flow, since it takes longer for the hot slug to reach the orbiter inlet. If the temperature limit is exceeded, the requirement is to revert to prevent the hot slug from traveling up the feedline where it could exceed saturation temperature and cause a geyser.

The estimated stop flow time limit is based on actual KSC stop flow data (magnitude of hot slug vs. time in stop flow), and is the estimated time to reach the temperature limit.

6.4.10.2 Reinitiation Following Revert

Pressure surges of about 200 psig are observed at the orbiter inlet pressure when the inboard fill and drain valve (PV10) is reopened following a revert. The cause is attributed to a GO_2 bubble, which develops in the orbiter fill line during the revert and collapses when PV10 is opened to reinitiate flow.

A requirement was added to ensure the orbiter inlet pressure was 85 to 100 psig for 2 minutes before opening PV10. This pressure was intended to ensure that PV10 had a slight flow through its relief feature to bleed out the gas trapped between the TSM vent line and PV10 in the uninsulated fill and drain line. This procedure worked at MSTL on T1-04 at a 2 percent liquid level. On STS-4, the procedure was used at 100 percent liquid level; however, it did not eliminate the surge pressure. Subsequent analysis showed that 120 to 130 psig was required at 100 percent to crack PV10. KSC objected to more than one pressure setting while RI indicated that the surge was not real since it was not noted at the orbiter pressure measurement between PV-9 and PV-10. The propellant loading committee agreed that further efforts should be dropped because the problem was deemed non-real.

6.4.10.3 Loss of SSME Bleed

If individual or total engine bleed flow is lost, it allows the engine heat leak of 10.25 Btu/sec per engine to enter the orbiter. For total loss of bleed, the math model predicted that saturation temperatures in the orbiter are reached in 12 to 20 minutes, depending on liquid level (2 percent to 100 percent), with no replenish flow. With replenish flow, this time could be delayed by several minutes (until the level reached the overfill sensor and resulted in an automatic stop flow). To prevent saturation limits from being reached, two sets of temperature limits are used to perform two different types of corrective action. The first limit is on engine inlet temperature, and ranges from -276° F at 100 percent to -286° F at 2 percent. If the initial engine inlet temperature is exceeded, the required corrective action is to stop flow and open the TSM vent. Opening the TSM vent drains the hot LO_2 from the orbiter and safes the system. If the TSM vent fails to open or if any engine inlet or feedline temperature approaches within 3° F of saturation, the requirement is to perform the emergency operations of closing the prevalves and pogo valves and initiating tank drain.

The initial limit on engine inlet temperature is set 5° F above the limit on orbiter inlet temperatures. Since engine inlet temperature generally runs 1° to 2° F warmer than orbiter inlet temperature, the initial limit on engine inlet temperature is 3° to 4° F above nominal. Since the limit on orbiter inlet ramp ranges from -291° F at 2 percent to -281° F at 100 percent, the initial limit on engine inlet temperature ranges from -286° F at 2 percent to -276° F at 100 percent.

6.4.11 Summary of Lessons Learned During Antigeysering System Development

The most difficult period relative to geysers is the initial fill to 2 percent period. During this time, the flight system is in a state of transient childown. Stable LO_2 overboard bleed flow through the engines has not been established, and warm slugs of fluid can be generated in the facility as a result of localized hot spots. With the relatively low hydrostatic head produced by the LO_2 in the ET, it is possible for saturation conditions to be reached within the fill/feed system as a result of these conditions. If saturation conditions are reached, a geyser can be generated. The effect of a geyser is to expel all, or a large part, of the fluid in the feedline. If the fluid level is within the tank, the voided line can be rapidly refilled. When the refill fluid encounters an obstruction, water hammer pressures of potentially destructive magnitude can be generated.

Special procedures were developed to prevent saturated conditions from occurring within the fill/feed system during the loading operation. The procedures have been progressively revised and refined throughout the flight program from STS-5 through STS-11 to overcome problems that appeared during various operations.

The initial procedure called for the inboard fill and drain (F&D) valve (PV10) to be closed during the orbiter chill period. The intent was to isolate the orbiter from the ground system after liquid has entered the ET 17-inch line. With PV10 closed, the LO_2 in the vehicle drains through the engine overboard bleed system. On the facility side, liquid drains out the TSM vent. The rationale for closing PV10 was to permit the vehicle to chill by evaporative cooling at relatively low pressure. This also permits higher pressures and consequently higher flows on the facility side to effect a more thorough childown of the facility lines and components.

For STS-6 FRF no. 1, the procedure described above was used. Shortly after PV10 was opened, a geyser occurred and a large portion of the ET feedline was voided. Fortunately, the LO_2 level had not reached the tank at the time of the geyser, and there was no refill water hammer. Figure 6.4-1 shows the dip in the engine inlet and manifold pressures resulting from the geyser. It was concluded that the geyser was caused by a warm slug of fluid that had been in the orbiter fill and drain line, which is bare. The pressure at the ORB/facility interface when PV10 was closed was sufficient to suppress boiling; therefore, the incoming heat to the F&D line went to bulk heating rather than to evaporation. The resulting warm slug of fluid reached saturation conditions as it was being transported through the ET 17-inch line and a geyser resulted. For the second STS-6 FRF and for STS-6 launch, the interface pressure was limited to 35 psig during the period when the inboard F&D valve was closed to reduce the heating effect on the fluid in the F&D line.

For STS-7, the closure of the inboard F&D valve during orbiter chill was abandoned and a revised procedure was employed for orbiter chill and fill to 2 percent.

After an initial facility flush period, LO_2 was pumped into the vehicle until the interface pressure increased to 34 psig. At 34 psig, the facility replenish valve was closed (stop flow), and the fluid in the vehicle drained back through the TSM vent and engine overboard bleed system. When the interface pressure decayed to 10 psig, the replenish valve was reopened and the TSM bleed was closed, thereby initiating the fill to 2 percent. Shortly after the fill to 2 percent started, a stop flow was commanded because the orbiter inlet temperature redline was violated. During the stop flow period for orbiter childown, hot slugs developed in the facility because of incomplete childown of the facility as well as several bare areas in the facility system.

As a result of the STS-7 event, the procedure was further modified to incorporate an additional 3-minute facility flush to eliminate the hot slugs created during the stop flow period. In addition, the bare areas of the facility system were insulated to reduce the total heat load. Per the new procedure, the orbiter inlet pressure is increased to 34 psig and then permitted to decay to 10 psig. The added flush is initiated when the orbiter inlet pressure decays to the 10 psig level by opening the replenish valve, the transfer line valve, and the TSM drain valve. The flush continues for 3 minutes, at which time the transfer line valve and the TSM drain valve are closed. With the closure of the transfer line valve and the TSM drain valve, fluid again enters the vehicle.

After the orbiter inlet pressure reaches 10 psig, the TSM vent is closed. Closure of the TSM vent is regarded as the start of the fill to 2 percent. After the initiation of fill to 2 percent, system temperatures are continually monitored to ensure that geysering conditions are avoided.

The revised procedure with the added facility flush was successfully used on STS-8, STS-9, and STS-11.

Table 6.4-II is a summary of system temperatures pertinent to avoiding geysering conditions from the fill to 2 percent through replenish. Although the values shown are for STS-9, they may be regarded as typical for vehicles with lightweight tanks. It is seen that all redlines were avoided with substantial margins.

In the event of a stop flow during the fill to 2 percent, a special problem is encountered. Heat from the facility convects into the open orbiter fill/feed system. Because of the low hydrostatic head of the fluid in the vehicle, the engine overboard bleed flow is relatively low during this period. As a consequence, the convected heat tends to collect in the orbiter 17-inch line. If flow into the vehicle was reinitiated with the warm slug of fluid in the 17-inch line, a geyser could be created when the fluid was transported into the ET 17-inch line. To avoid such an occurrence, a special procedure has been added for reinitiation after a stop flow below the 2 percent level. The special procedure consists of opening the TSM vent for 4 minutes to permit the warm slug to be flushed overboard. At the end of the 4-minute drain period, the replenish valve is opened to the 12 percent position, and the TSM vent is left open; this condition is held for 2 minutes. At the end of the 2-minute period, the system redline values are monitored. If the limiting values are not exceeded, the replenish valve goes full open and the TSM vent is closed, thereby completing the reinitiation of the fill to 2 percent. This procedure was successfully demonstrated during the STS-9 tanking tests.

On the basis of the successful performance of the loading operations for STS-8, STS-9, and STS-11, it can be concluded that the procedures for geyser avoidance are satisfactory.

6.4.12 Rockwell Assessment of Antigeysering Procedure for Lightweight Tanks

The most difficult period relative to geysers is the initial fill to 2 percent period. During this period, the flight vehicle system is in the state of transient chilldown, stable LO_2 overboard bleed flow through the engines has not been established, and warm slugs of fluid can be generated in the facility as a result of localized hot spots. With the relatively low hydrostatic head produced by the LO_2 in the ET, it is possible for saturation conditions to be reached within the fill/feed system as a result of the aforementioned conditions. If saturation conditions are reached, a geyser can be generated. The affect of a geyser is to expel all, or a large part, of the fluid in the feedline. If the fluid level is within the tank, the voided line can be refilled rapidly. When the refill fluid encounters an obstruction, water hammer pressures of potentially destructive magnitude can be generated.

Special procedures were developed to prevent saturated conditions from occurring within the fill/feed system during the loading operation. The procedures have been revised and refined progressively throughout the flight program from STS-5 through STS-11 to overcome problems that appeared during the various operations.

The initial procedure called for the inboard fill and drain valve (PV10) to be closed during the orbiter chill period. The intent was to isolate the orbiter from the ground system after liquid had been loaded to part way up the ET 17-inch line. With PV10 closed, the LO_2 in the vehicle drained through the engine overboard bleed system while liquid was pumped out the tail service mast vent on the facility side. The rationale for closing PV10 was to permit the vehicle to chill by evaporative cooling at relatively low pressure, yet permit higher pressures and consequently higher flows on the facility side. This effected a more thorough chilldown of the facility lines and components.

For STS-6 FRF no. 1, the procedure just described was used. Shortly after PV10 was opened, a geyser occurred and a large portion of the ET feedline was voided. Fortunately, the LO_2 level had not reached the tank at the time of the geyser and there was no refill water hammer. Figure 6.4-1 shows the dip in the engine inlet and manifold pressures resulting from the geyser. It was concluded that the geyser was caused by a warm slug of fluid that had been in the orbiter fill and drain line, which is bare. The pressure at the ORB/facility interface when PV10 was closed was sufficient to suppress boiling; therefore, the incoming heat to the F&D line went to bulk heating rather than to evaporation. The resulting warm slug of fluid reached saturation conditions as it was being transported through the ET 17-inch line and caused a geyser. For the second STS-6 FRF and for STS-6 launch, the interface pressure was limited to 35 psig during the period when the inboard F&D valve was closed to reduce the heating affect on the fluid in the F&D line.

For STS-7, the closure of the inboard F&D valve during orbiter chill was abandoned and a revised procedure was employed for orbiter chill and fill to 2 percent. After an initial facility flush period, LO_2 was admitted into the vehicle until the interface pressure increased to 34 psig. At 34 psig, the facility replenish valve was closed (stop flow) and the fluid in the vehicle drained back through the TSM vent plus the engine overboard bleed system. When the interface pressure decayed to 10 psig, the replenish valve was reopened and the TSM bleed was closed, thereby initiating the fill to 2 percent. Shortly after the fill to 2 percent started, a stop flow was commanded because the orbiter inlet temperature redline was violated. During the stop flow period for orbiter chilldown, hot slugs developed in the facility as a result of incomplete chilldown of the facility, plus several bare areas in the facility system.

Because of the STS-7 event, the procedure was further modified to incorporate an additional 3-minute facility flush to eliminate the hot slugs created during the stop flow period. In addition, the bare areas of the faility system were insulated to reduce the total heat load. For the new procedure, the orbiter inlet pressure is increased to 34 psig and then permitted to decay to 10 psig, as described earlier, by flow through the TSM vent and the engine overboard bleed system. For the revised procedure, the added flush is initiated when the orbiter inlet pressure decays to the 10 psig level. For the flush, the replenish valve, the transfer line valve, and the TSM drain valve are opened. The flush continues for 3 minutes, at which time the transfer line valve and the TSM drain valve are closed. With the closure of the transfer line valve and the TSM drain valve, fluid again enters the vehicle.

After the orbiter inlet pressure reaches 10 psig, the TSM vent is closed. Closure of the TSM vent is regarded as the start of the fill to 2 percent. After the initiation of fill to 2 percent, system temperatures are continuously monitored to ensure that geysering conditions are avoided.

The revised procedure with the added facility flush was successfully used on STS-8, STS-9, and STS-11. All redline values were observed with satisfactory margins. Figure 6.4-2 shows the orbiter inlet pressure for

STS-9, which can be regarded as a typical operation for the new procedure. The various valve operations are noted on figure 6.4-2 for the orbiter chill and fill plus the fill to 2 percent phases. Figures 6.4-3 through 6.4-7 show the critical parameters during the fill to 2 percent for STS-9 and also show that the redline limits were observed. Figure 6.4-3 shows that the orbiter inlet temperature briefly exceeded the redline value at approximately T - 240 seconds as a result of a TSM burp. The TSM burb is a common occurrence that is caused by stagnant LO_2 in the uninsulated TSM vent line. The exceedance did not continue longer than 30 seconds, which is the limit for the orbiter inlet temperature during fill to 2 percent. All other parameters have a 20-second limit for exceedances before the implementation of corrective actions.

Table 6.4-II (p. 6.4-10) is a summary of system temperatures pertinent to avoiding geysering conditions from the fill to 2 percent through replenish. Although the values shown are for STS-9, they may be regarded as typical for vehicles with lightweight tanks. It is seen that all redlines were avoided with substantial margins.

In the event of a stop flow during the fill to 2 percent, a special problem is encountered. Heat from the facility convects into the open orbiter fill/feed system. Because of the low hydrostatic head of the fluid in the vehicle, the engine overboard bleed flow is relatively low during this period. As a consequence, the convected heat tends to collect in the orbiter 17-inch line. If flow into the vehicle were reinitiated with the warm slug of fluid in the 17-inch line, a geyser would be created when the fluid was transported in the ET 17-inch line. To avoid such an occurrence, a special procedure has been added for reinitiation after a stop flow below the 2 percent level. The special procedure consists of opening the TSM vent for 4 minutes to permit the warm slug to be flushed overboard. At the end of the 4-minute drain period, the replenish valve is opened to the 12 percent position and the TSM vent is left open, a condition that is held for 2 minutes. At the end of the 2-minute period, the system redline values are monitored. If the limiting values are not exceeded, the replenish valve goes full open and the TSM vent is closed, thereby completing the reinitiation of the fill to 2 percent. This procedure was successfully demonstrated during the STS-9 tanking tests.

On the basis of the successful performance of the loading operations for STS-8, STS-9, and STS-11, it can be concluded that the procedures for geyser avoidance are satisfactory. It is intended that these procedures be used for the duration of the program, although some variation in the flush procedure is required for VAFB because of differences on the facility plumbing. The geyser avoidance performance at VAFB should be at least as good as experienced at KSC.

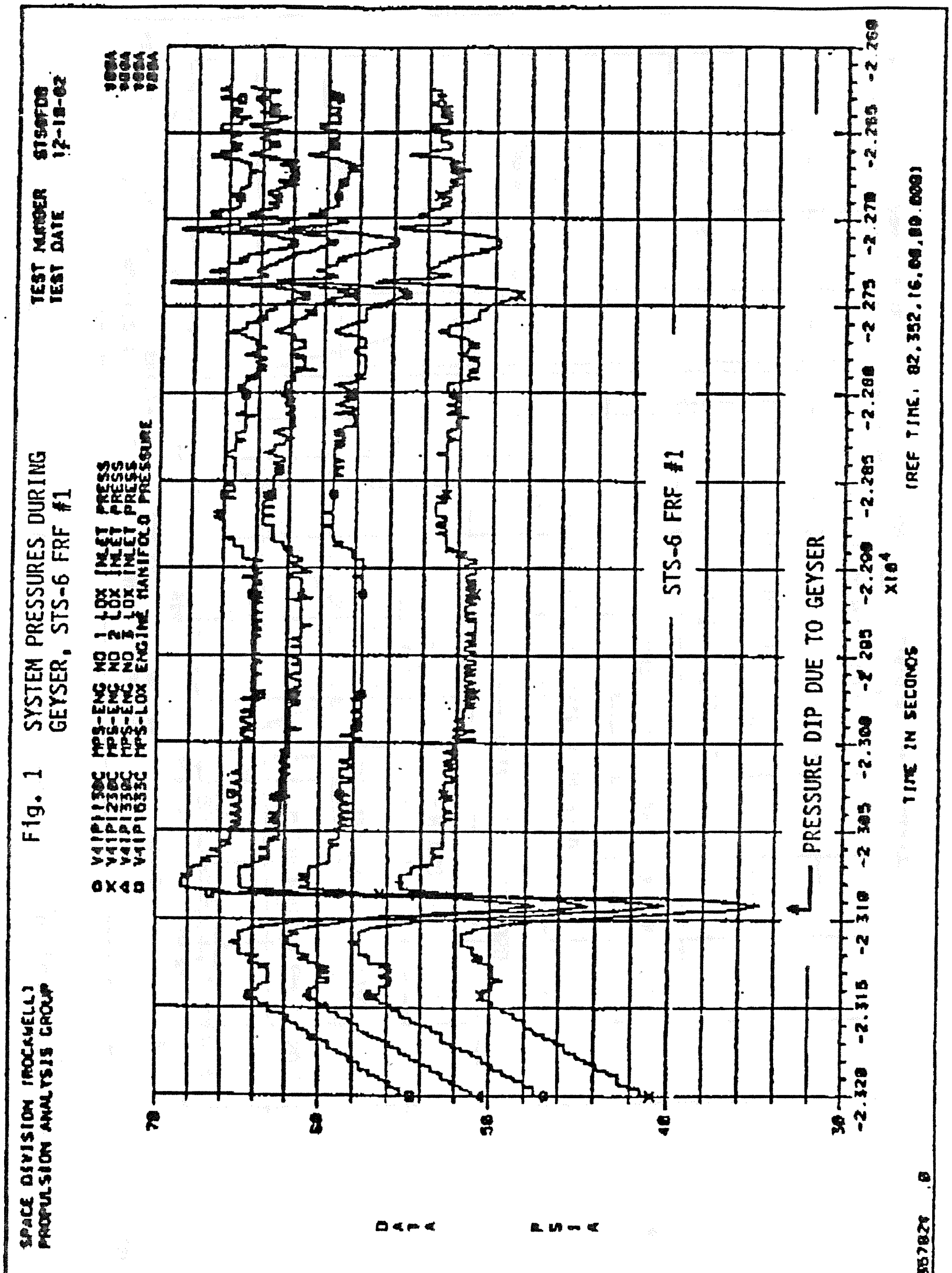


Figure 6.4-1.- System pressures during geyser, STS-6 FRF no. 1.

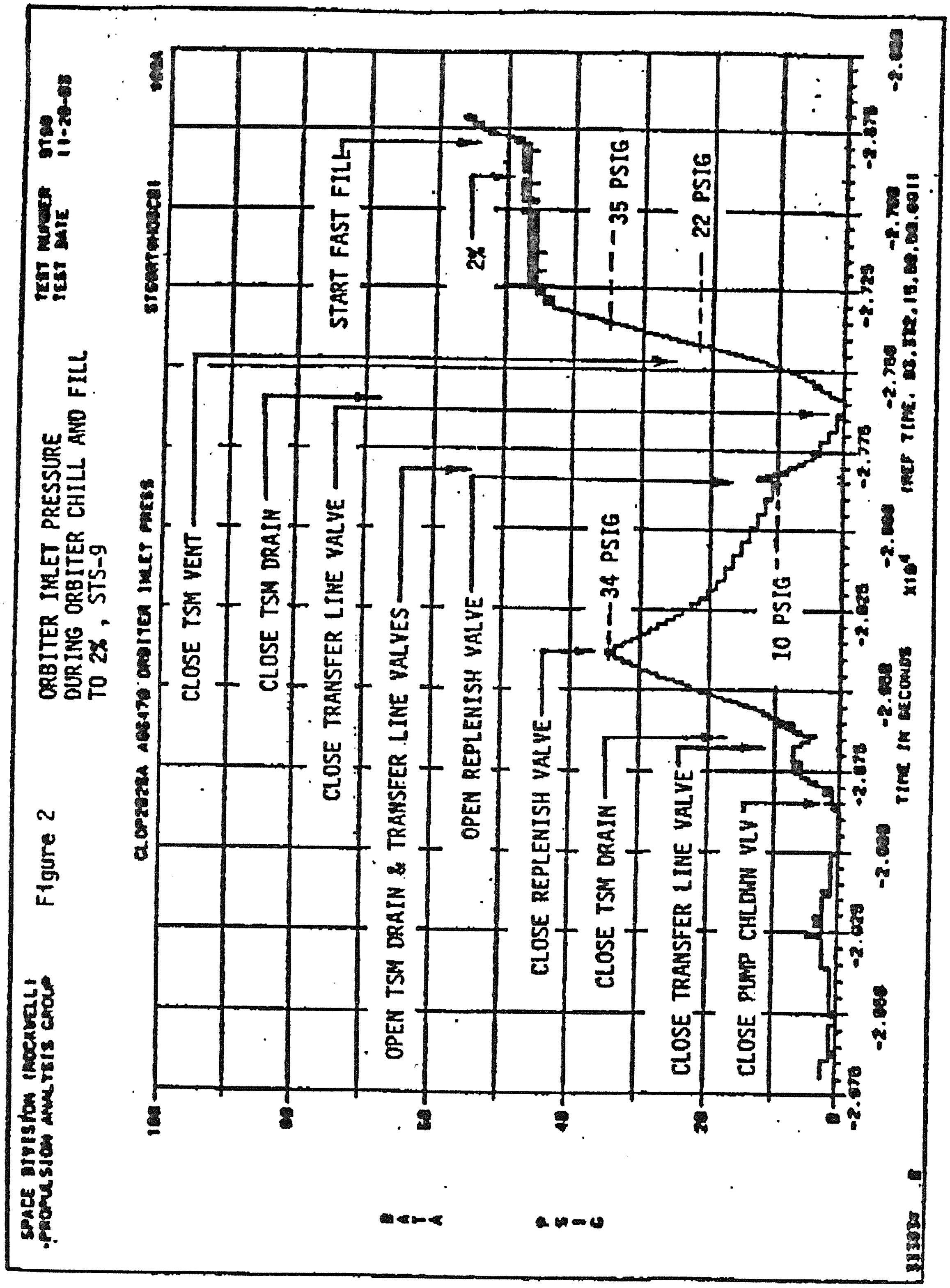


Figure 6.4-2.- Orbiter inlet pressure during chill and fill to 2 percent, STS-9.

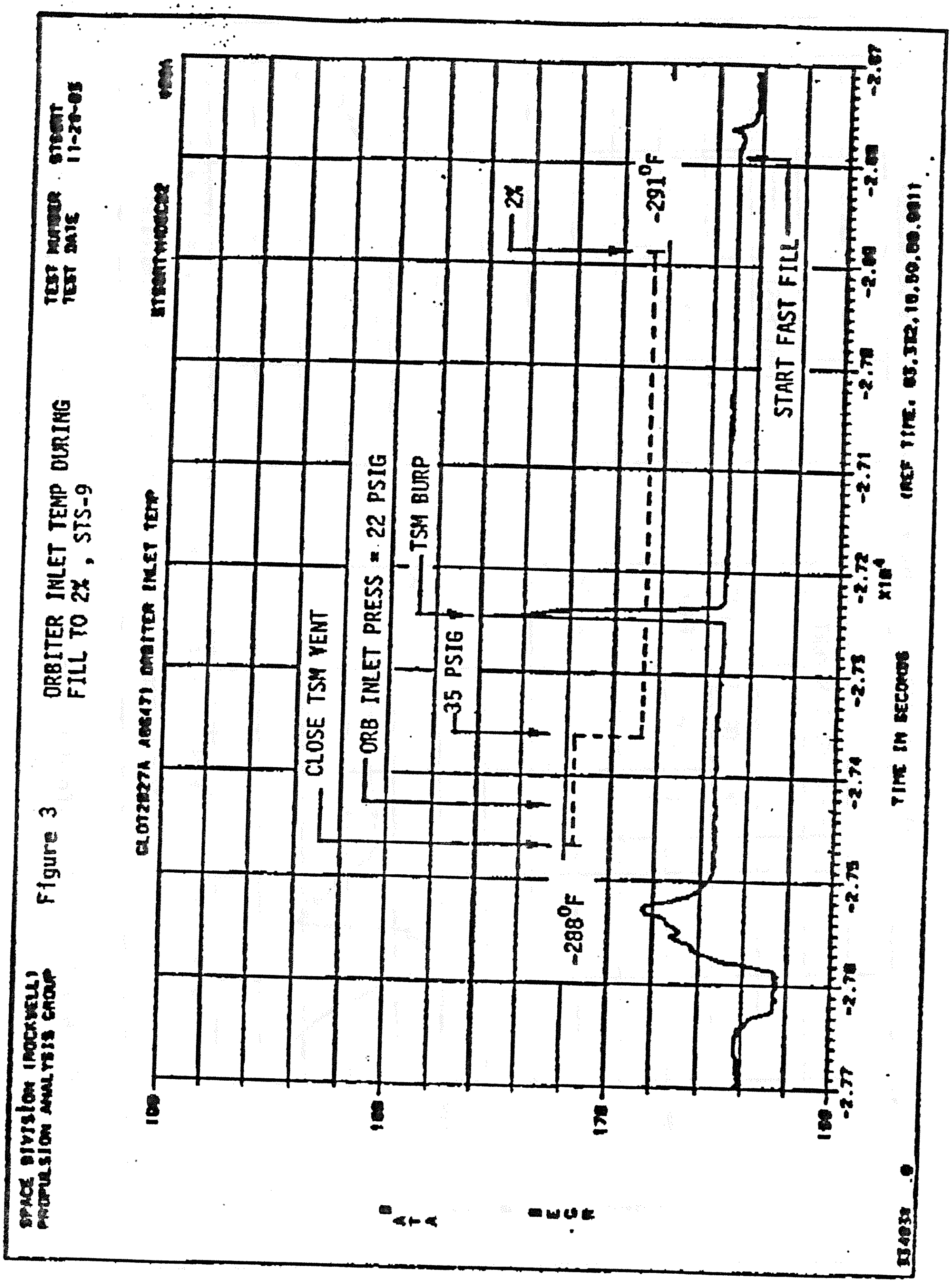


Figure 6.4-3.- Orbiter inlet temperature during fill to 2 percent, STS-9.

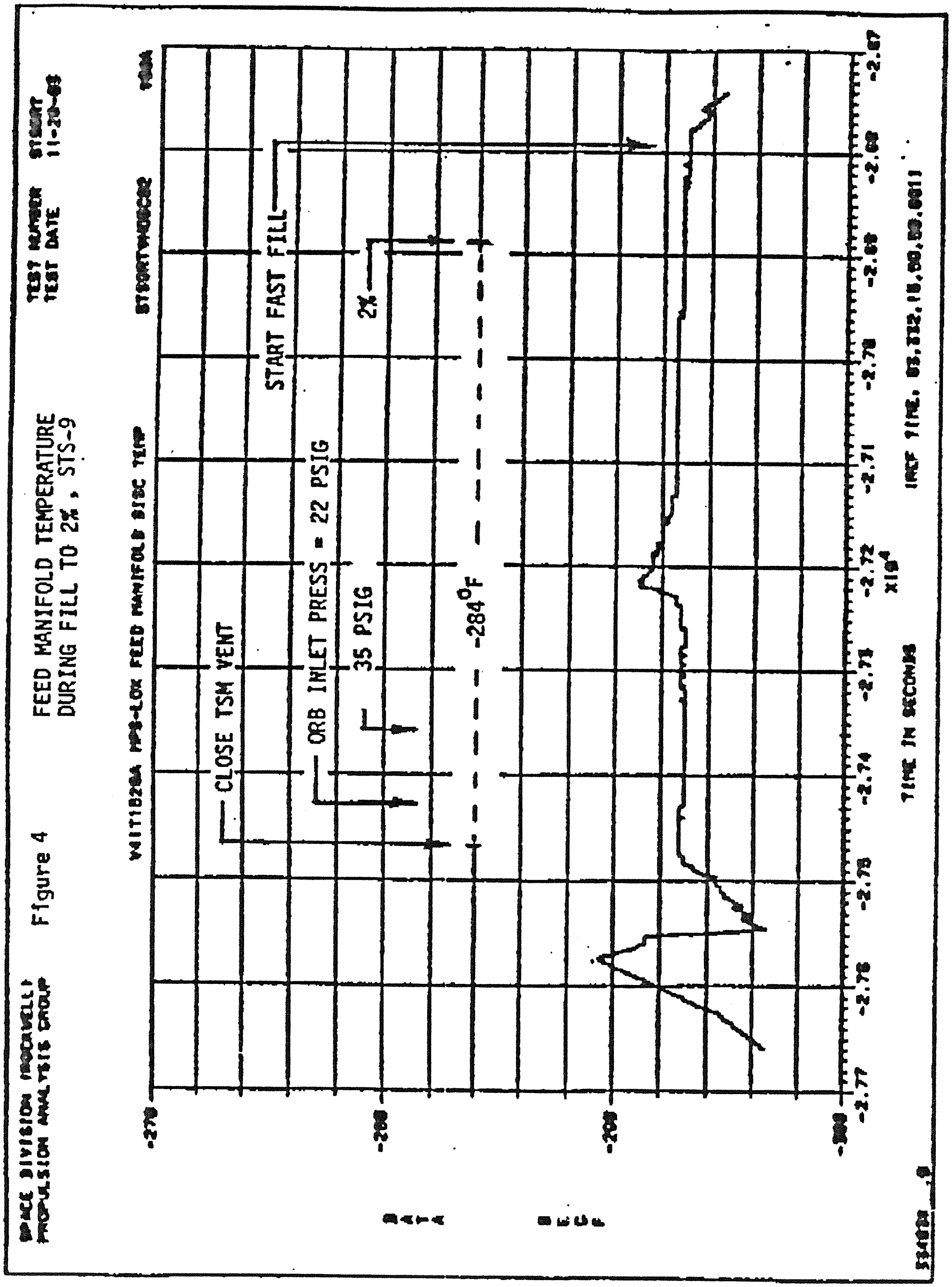


Figure 6.4-4.- Feed manifold temperature during fill to 2 percent, STS-9.

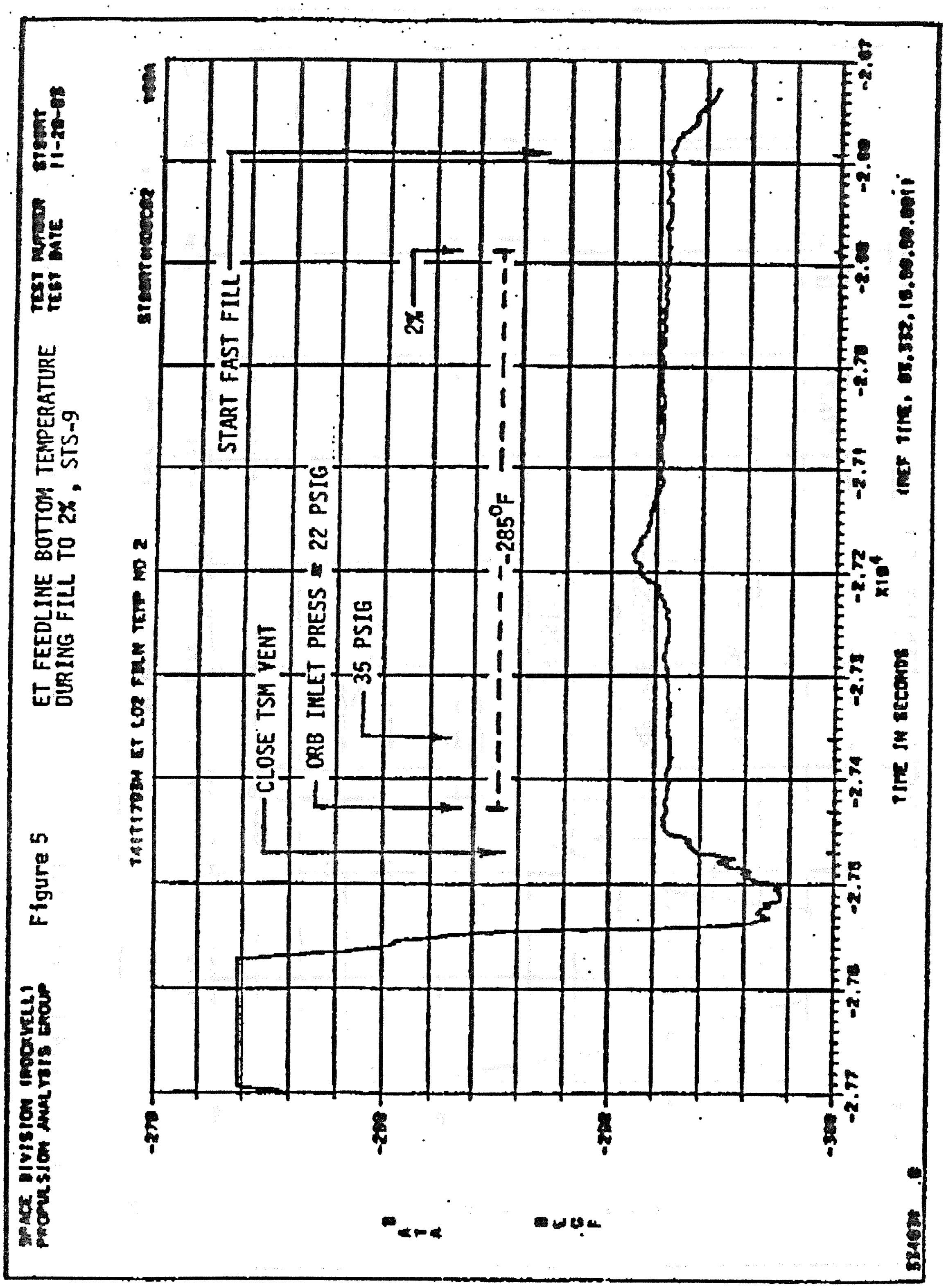


Figure 6.4-5.- ET feedline bottom temperature during fill to 2 percent, STS-9.

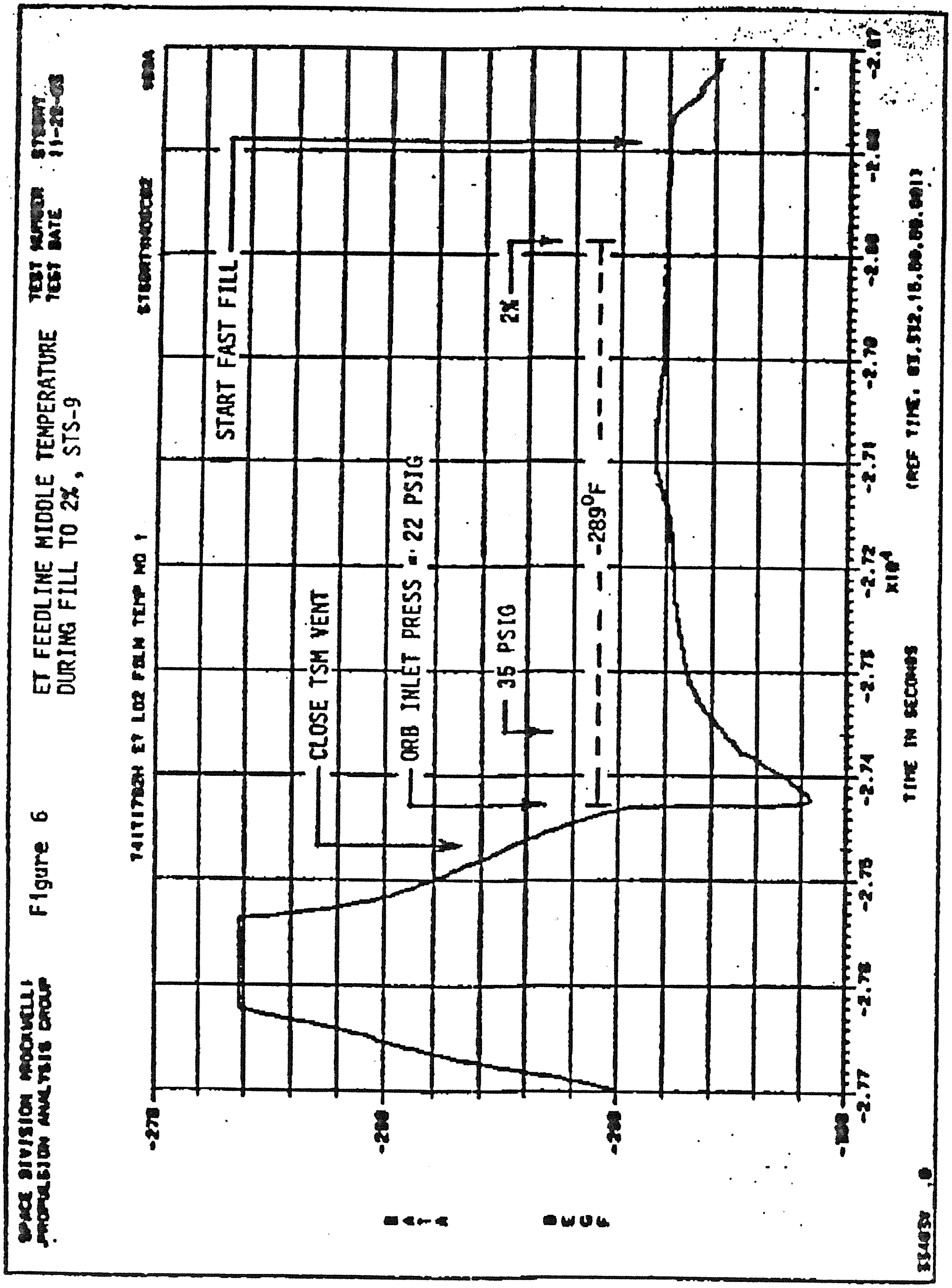


Figure 6.4-6.- ET feedline middle temperature during fill to 2 percent, STS-9.

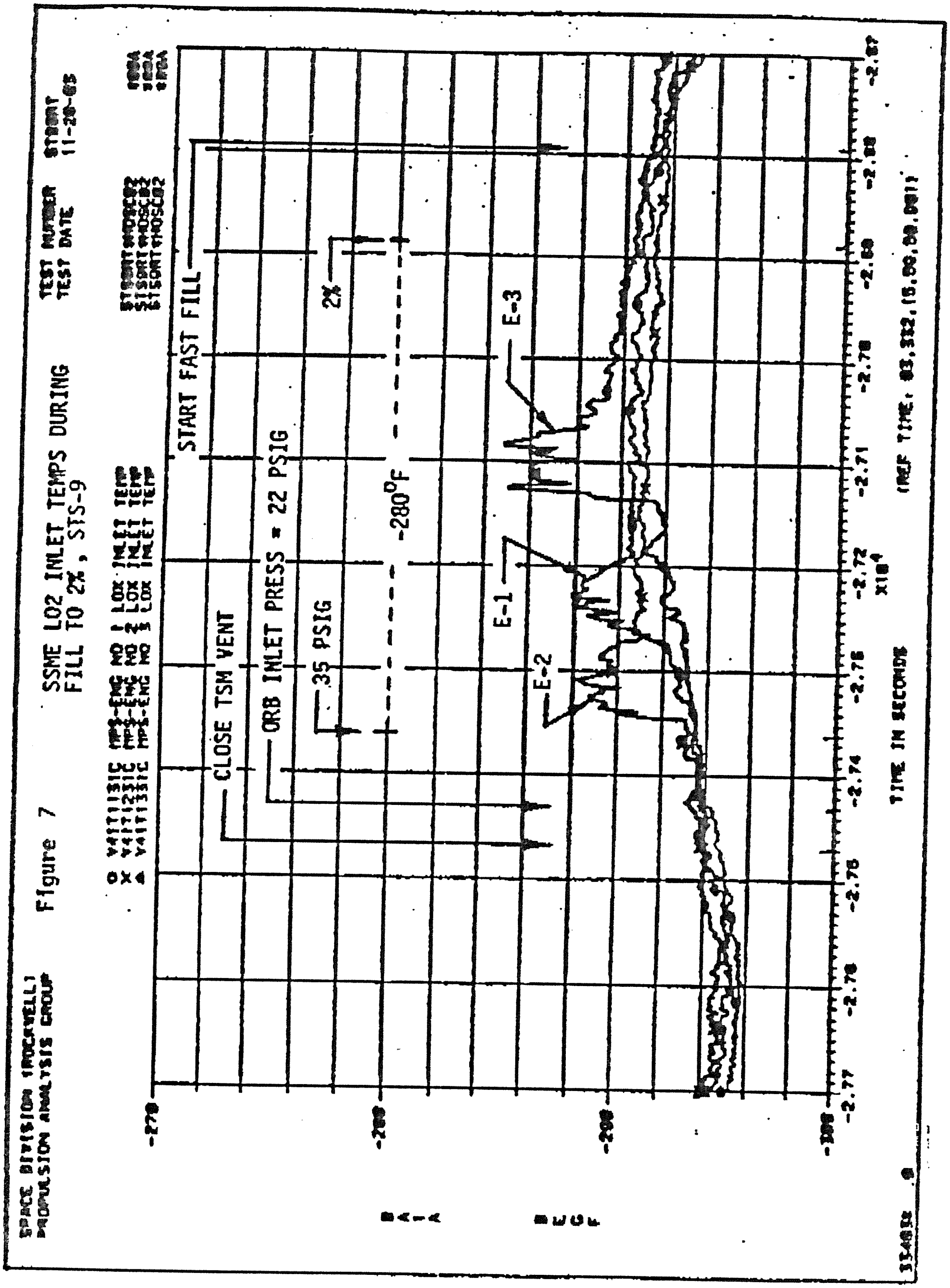


Figure 6.4-7.- SSME LO2 inlet temperatures during fill to 2 percent, STS-9.

7.1 LIQUID ROCKET FUNDAMENTALS

7.1.1 Purpose and Scope

The purpose of this systems brief is to provide a basic background of the history, theory, optimization, design, and operation of liquid rocket engines. This section was written principally as an introductory guide for new Booster Systems engineers who have a basic college background in physics, thermodynamics, and compressible fluid flow. The material covered involves not specifically the space shuttle liquid rocket engines, but rather involves concepts and designs which apply to all liquid rocket propulsion systems.

Liquid rocket engines have a wide area of applications, including Earth to low orbit or Earth escape trajectories, Earth orbital transfers, lunar and planetary transfers, lunar and planetary landers, station keeping and attitude control, and rendezvous and docking maneuvers.

A separate section on solid rocket motor fundamentals, not strictly limited to those of the space shuttle design, is provided in systems brief 7.5. High pressure gases can also be stored in tanks for propellant usage, such as nitrogen gas which is used on the space shuttle manned maneuvering unit (MMU). This systems brief will not cover gaseous propellant storage systems.

7.1.2 History

The earliest records of rocket use date back to the 13th century in ancient China where solid propellant rockets were used in the siege of Kaifeng in A.D. 1232. The defenders of the city used the rockets in a similar way to the use of fire arrows. These rockets used a solid black powder charge for their propulsion consisting of gunpowder and additional charcoal to slow the burn rate. The invention quickly spread and solid rockets were used in Europe in 1258 in Cologne in what is present day Germany. The development of the rocket charges led to the invention of the gun almost a century later in the 1300's.

Liquid rocket propulsion was developed much later than solid rockets and did not appear until early in this century. Soon after 1900 a Russian schoolteacher, Konstantin Tsiolkovsky, began to write about rockets and developed some of the basic theories and ideas on rocket travel. He believed large rockets could best be built by using liquid rather than solid propellants. Unfortunately, he did not experiment with any of his ideas.

The first person to actually design and build a liquid rocket was American physicist Robert Goddard (1882-1945). He began working with liquid propellants in 1923 and had a gasoline and liquid oxygen motor running on a test stand in November 1923. His first liquid launched rocket fired from Auburn, Mass. on March 16, 1926, traveled in a short arc and impacted the ground 2.5 seconds after liftoff, having covered 184 feet. Some of his liquid rockets reached 7500 feet altitude in the 1930's. American interest on rockets lay dormant through most of the 1930's, however, and really did not advance until after World War II.

A German group consisting of Johannes Winkler, mathematician Hermann Oberth, engineers Rudolf Nebel and Klaus Riedel and an engineering student, Wernher von Braun, built and experimented with a similar liquid oxygen and gasoline rocket in 1930. They first launched experimental rockets a year later, in 1931.

The German army decided to evaluate the potential of rockets as weapons beginning in 1932. This effort culminated in the first successful flight of a V-2 rocket (Vergeltungswaffe -2) or Vengeance Weapon 2, in October 1942. These rockets, each carrying a 2200 lb warhead, were not used as operational weapons against the allied forces till September 1944 in the introduction of attacks against the British. The V-2 rockets were fueled by a 75 percent grain alcohol/25 percent water mixture and liquid oxygen. The propellants were fed into a combustion chamber at 320 psia by centrifugal pumps which were driven by hydrogen peroxide driven steam turbines. Steering was accomplished by a combination of external rocket fins and four graphite vanes in the exhaust flow. The record altitude of this rocket during the V-2 program was 114 miles.

After World War II ended, the USA developed a rocket technology and research program at White Sands Proving Ground in New Mexico. The first successor to the V-2 was the United States Navy's Viking, which stood about the same height as the V-2 (46 feet) but was slimmer and had a liftoff weight of 9650 lb. The Viking was very similar to the V-2 with the exception of steering control. The rocket nozzle was attached to a gimbal ring that could tilt rather than use exhaust vanes.

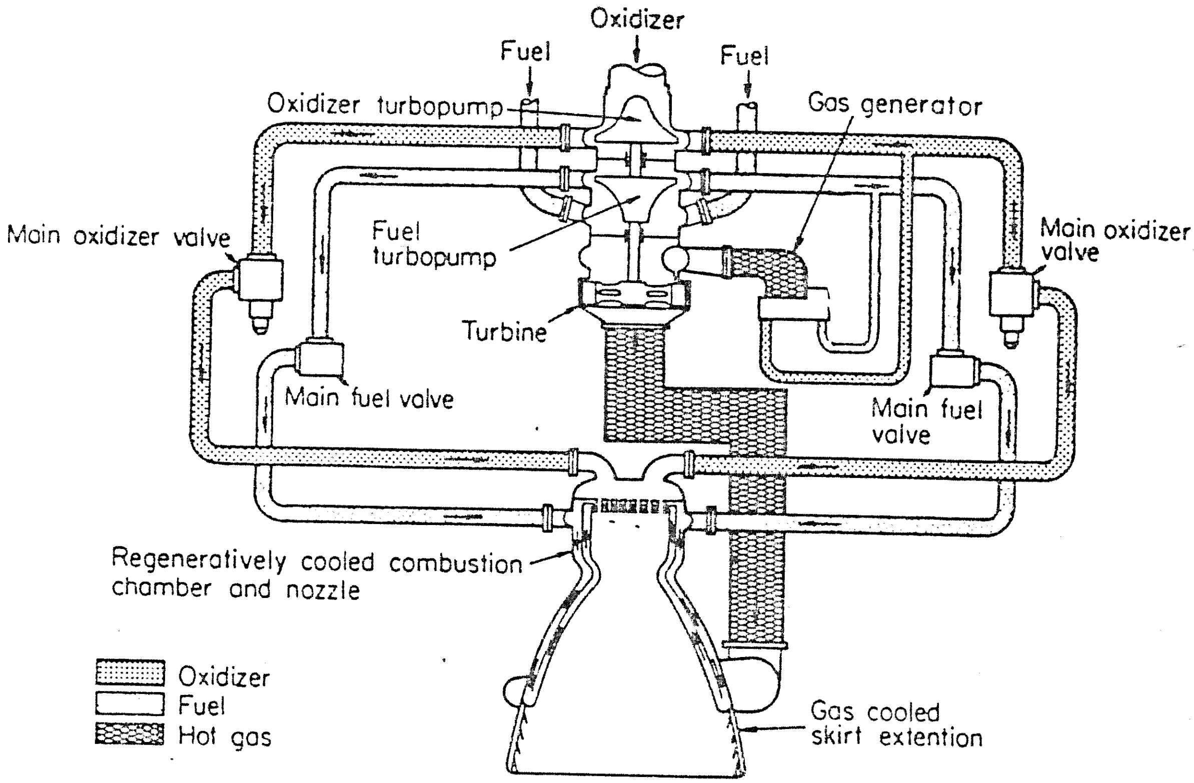
Multiple stage rockets were first developed with the initiation of Project Bumper in the United States in the late 1940's. Bumper was a two-stage rocket devoted to high altitude research studies, consisting of a first stage V-2 (also known as the A-4) and a second stage WAC Corporal. The WAC Corporal was an 11-foot rocket powered by pressure-fed tanks of nitric acid and aniline. On February 24, 1949 a Bumper upper stage reached a record altitude of 250 miles. This record altitude existed until 1956 when it was surpassed by the Army Jupiter C rocket that reached about 500 miles altitude. In 1957, with the initiation of a daring United States program called Project Farside, a four-stage solid rocket was launched from a balloon at a height of 80,000 feet and reached an altitude of at least 3000 miles.

During this same time period in the late 1930's through the 1950's the Soviet Union was also experimenting heavily with high altitude sounding rockets. Their research culminated with the launch of the first satellite in low Earth orbit, Sputnik 1, on October 4, 1957. This satellite was launched by a liquid oxygen/kerosene derivative of one of their intercontinental ballistic missiles. Sputnik 2, the first satellite to carry a live animal, a dog, was launched in November 1957. The first United States satellite, Explorer 1, was launched aboard an upgraded Jupiter C rocket on January 31, 1958. In October of that year, the United States established NASA to carry out this nation's civilian space program. The following year, 1959, the Soviets launched the Luna 3 spacecraft, which was the first craft to photograph the far side of the moon. The United States launched Surveyor spacecraft capable of soft landing on the lunar surface using retrorockets in the 1960's.

The Soviet Union launched the first man into space, Yury Gagarin, aboard a Vostok capsule on April 12, 1961. The United States followed with the launch of Alan Shepard on May 5, 1961 aboard a Mercury capsule. Gemini and Voskhod programs in the mid 1960's provided enhanced manned launch capability. France became the third nation to launch a satellite in 1965.

The United States first succeeded in landing men on the moon on July 20, 1969 with the launch of the Apollo 11 vehicle aboard a Saturn V rocket with the three man crew of Neil Armstrong, Buzz Aldrin, and Michael Collins. This vehicle stood 363 feet tall and weighed a massive 6,000,000 lb at liftoff. It was capable of placing 230,000 lb in low Earth orbit or taking 95,000 lb to escape velocity. The Saturn V vehicle used five F-1 liquid oxygen/kerosene first stage engines, each of which produced 1,500,000 lb thrust and burned for 2.5 minutes (fig. 7.1-1). At about 36 miles altitude, the second stage engines were ignited. The second stage used five J-2 engines that burned liquid oxygen and liquid hydrogen. The third stage ignited about 9 minutes into flight and used a single J-2 engine.

The Apollo service module, used to transport the crew from the Earth to Moon orbit and back, used a single 22,000 lb thrust pressure-fed engine using nitrogen tetroxide and hydrazine (fig. 7.1-2). A two-stage craft carried aboard the Apollo third stage with the service module, the lunar module, was used to carry two astronauts between lunar orbit and the Moon's surface. It also used pressure-fed nitrogen tetroxide and hydrazine engines for both stages. The first stage of the lunar module was used to descend to the Moon's surface, while the second stage used the resting first stage as a launch pad to blast off into lunar orbit with the crew a few days later (fig. 7.1-3). A total of six successful lunar landings and one aborted Apollo voyage (Apollo 13) were made during the Apollo program. After the conclusion of the Moon landing program, a few remaining Saturn rockets were used to launch the Skylab large space station and three sets of crew to carry out long duration space studies. The Soviets also carried out long duration space missions aboard their own space stations.



Propellants:	LO_2 and RP-1 (kerosene)
Power:	Gas generator/turbopump
Specific impulse:	265 sea level/305 vacuum
Chamber pressure:	1122 psia
Thrust:	1,522,000 lb (sea level)
Weight:	18,616 lb
Throttling:	None available

Figure 7.1-1.- Saturn launch vehicle first stage F-1 engine propellant flow.

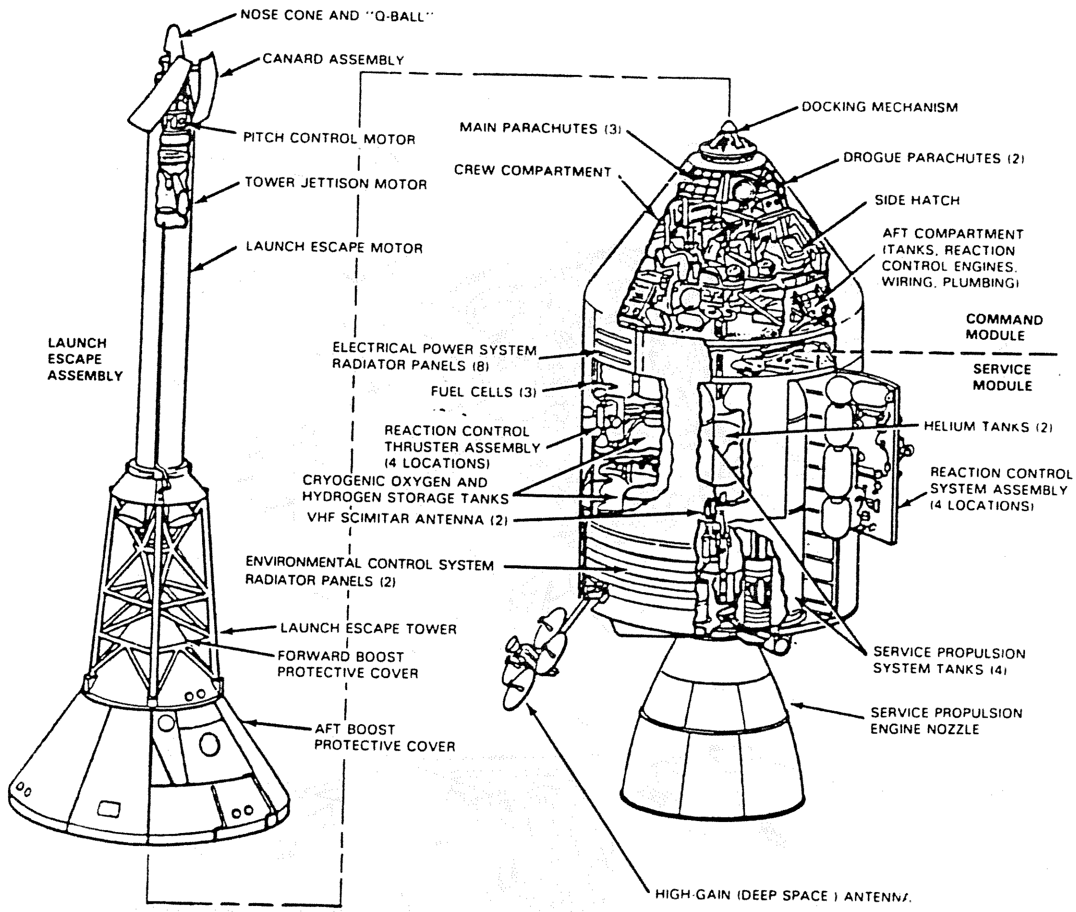


Figure 7.1-2.- Apollo service and command module with escape tower.

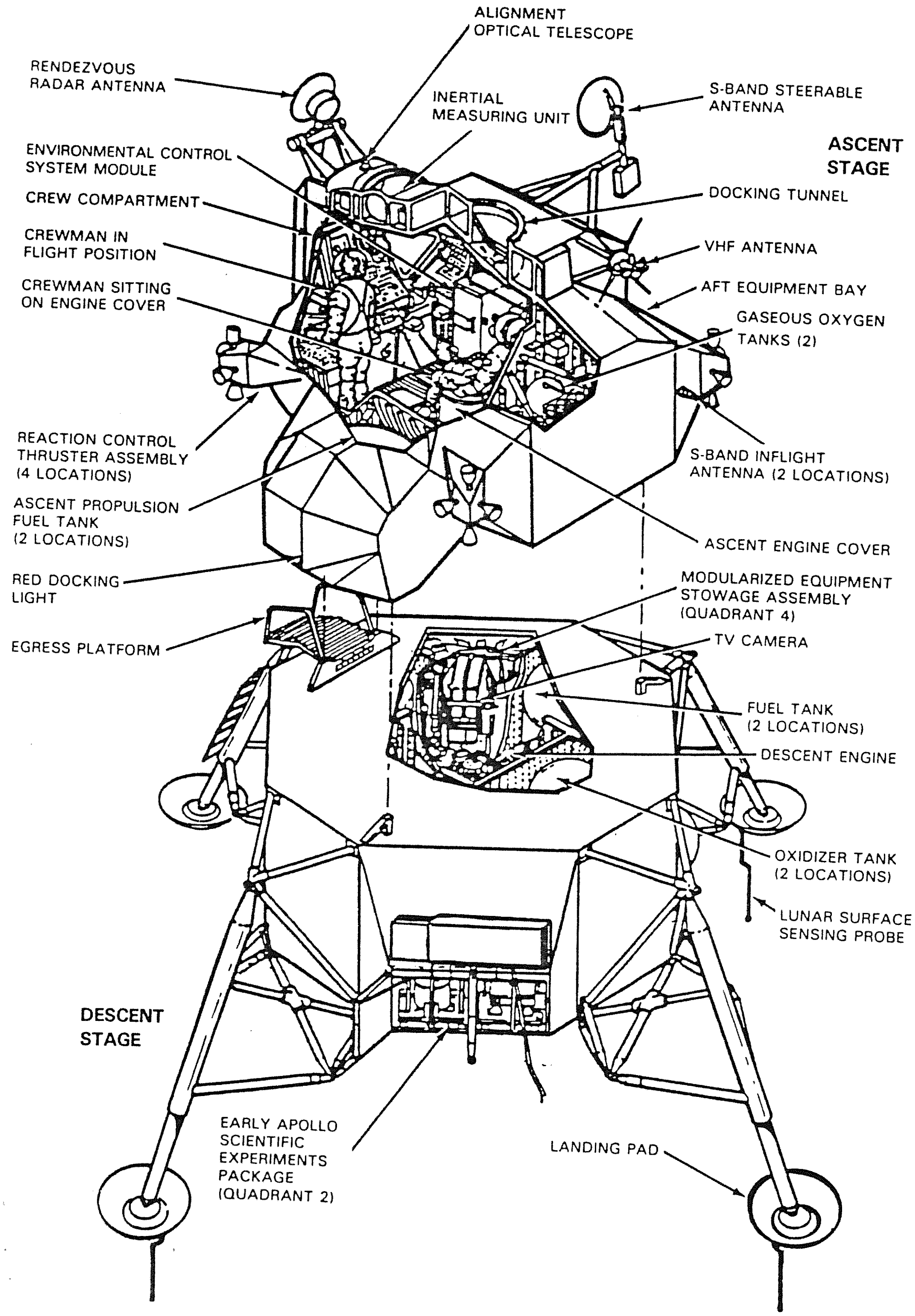


Figure 7.1-3.- Lunar ascent and descent stages.

Other nations rapidly developed their own space capabilities during this same time period. Satellite launches were introduced by Japan (1970), China (1970), and Britain (1971).

The European Space Agency (ESA), whose efforts are jointly conducted by several western European nations, principally France and Germany, established a launch site near Kourou, French Guinea. Ariane rockets are launched from this site, taking advantage of the Earth's rotational speed to lower propellant quantities needs to achieve orbit.

The only major cooperative United States and Soviet manned space mission occurred with the Apollo-Soyuz project in 1975. A three-man Apollo crew and a two-man Soviet crew rendezvoused in orbit and linked up with the aide of a docking module. The two crews carried out an Earth survey, space science, and biomedical experiments in orbit.

Exploration programs of the planets were also initiated by both the Soviets and Americans in the 1960's and 1970's. Long range spacecraft requiring large velocities and precise navigation were sent to Venus, Mars, Mercury, and Jupiter. Two American Viking spacecraft made successful soft landings on Mars in 1976 with a combination of liquid fueled retrorockets and parachute assist (fig. 7.1-4a). The United States Voyager spacecraft made encounters with Saturn, Uranus, and Neptune in the 1980's (fig. 7.1-4b). The required lifetimes and reliability of the small liquid rocket engines to guide the deep space probes are immense. In the case of Voyager 2, its encounter with Neptune in August 1989 came almost 12 years after its launch.

In a switch of policy in the 1980's, the United States decided to base its manned space program on reusable spacecraft with the shuttle program rather than expendable vehicles as had been used in the Mercury, Gemini, and Apollo programs. The shuttle vehicle stack consists of a reusable orbiter powered by three liquid hydrogen and liquid oxygen engines, each producing about 470,000 lb vacuum thrust, and two strap-on solid propellant boosters. The cryogenic propellants are carried in an expendable tank which is jettisoned on-orbit. The two solid boosters burn during the first 2 minutes of flight and are then jettisoned for a parachute recovery in the Atlantic Ocean. The first flight of the United States shuttle was in April 1981. The United States shuttle can lift off with a crew of seven and about 55,000 lb of payload in its standard configuration and meet requirements for a safe abort landing if an engine should shut down early in flight. A view of several major United States launch vehicles used in the 1970's and 1980's is shown in figure 7.1-5.

The Soviet Union also developed a space shuttle that could be launched aboard a heavy lift Energiya vehicle. It is similar to the United States version with the exceptions that all liquid propulsion system engines are used during ascent and expendable engines attached to the external tank are jettisoned once near-orbital velocity has been reached. The Energiya launch vehicle of the Soviet shuttle uses strap-on kerosene and liquid oxygen engines coupled with a longer burning core of liquid hydrogen and oxygen engines. The Soviet orbital maneuvering engines use liquid oxygen and kerosene as opposed to the nitrogen tetroxide and monomethylhydrazine the

United States orbiter uses. The Soviet shuttle was launched in its maiden flight, in an unmanned configuration, in November 1988. The Soviets also keep an expendable manned launch vehicle system active as a backup to their shuttle launch system to provide greater assurance of access into space.

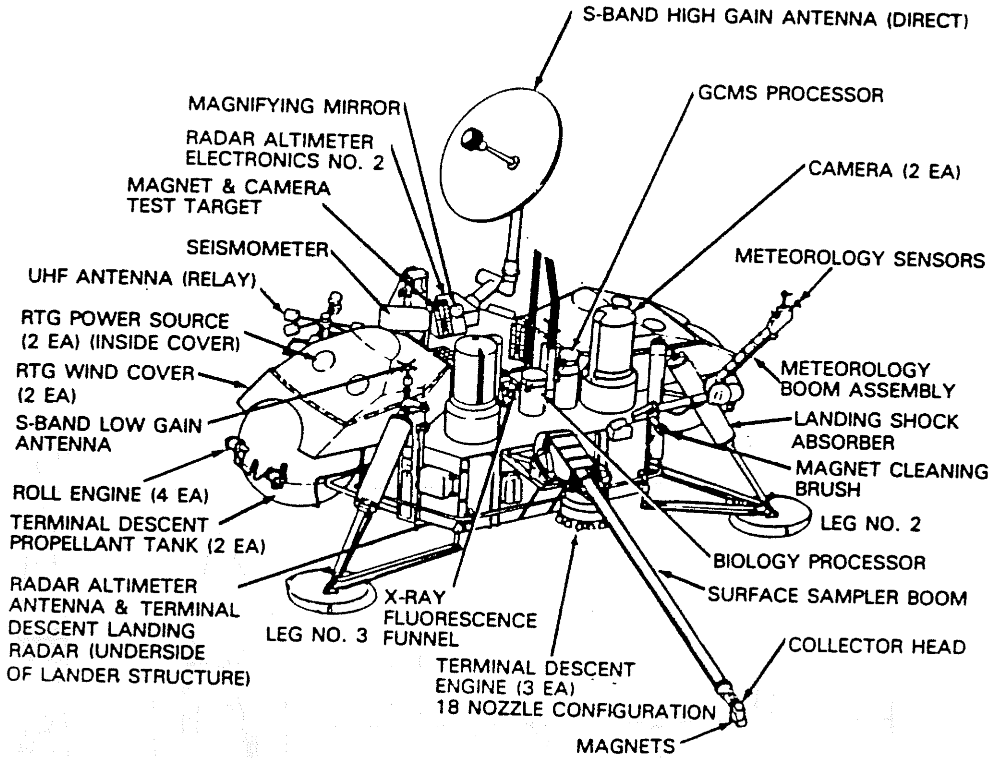


Figure 7.1-4a.- Viking lander.

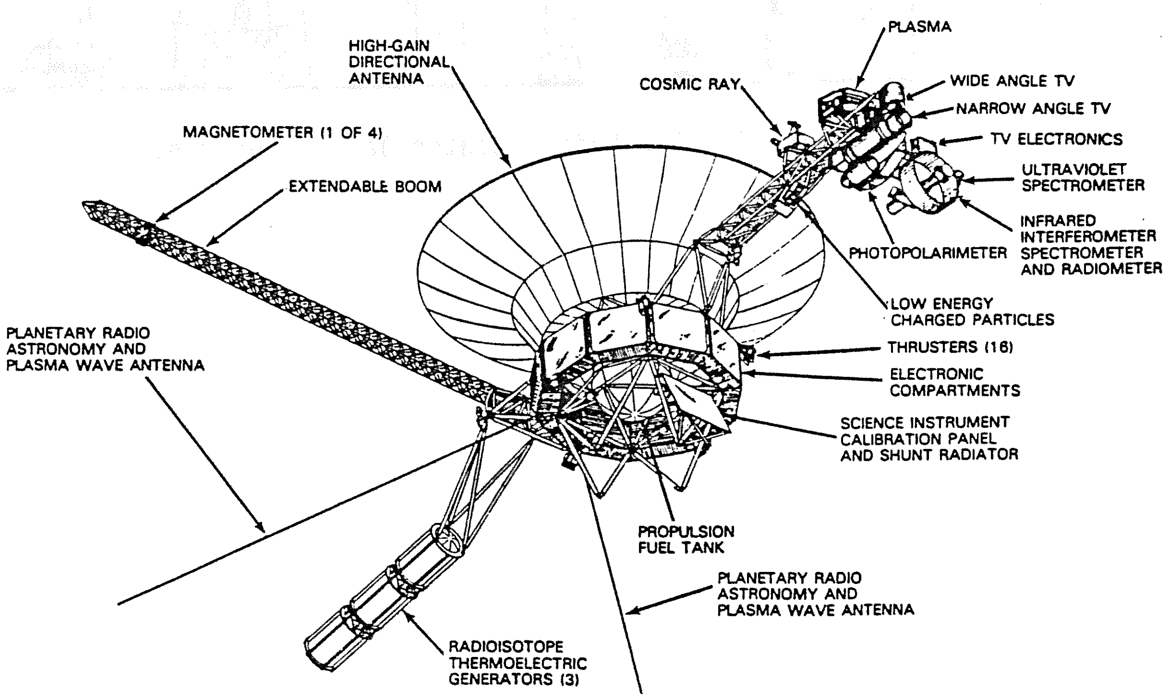


Figure 7.1-4b.- Voyager spacecraft.

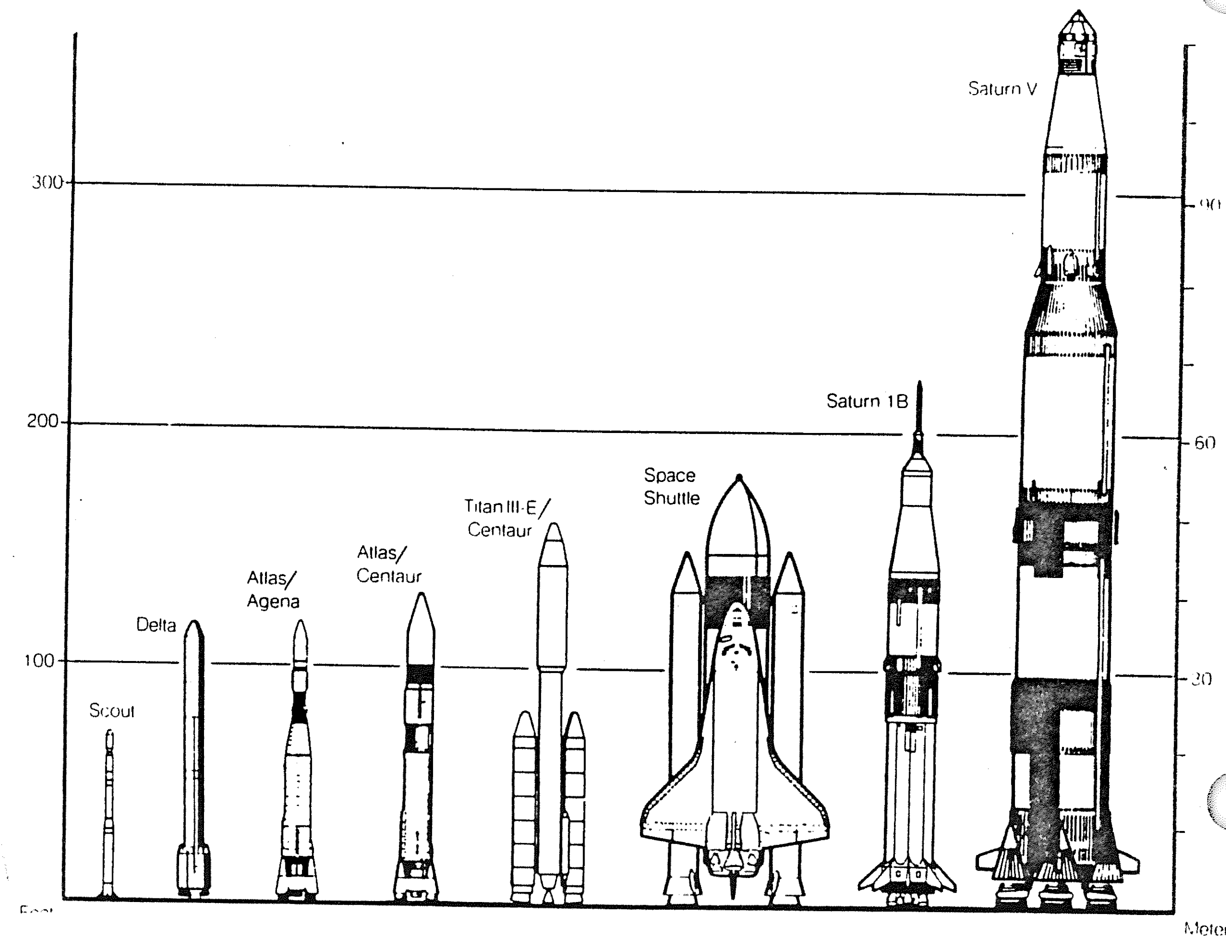


Figure 7.1-5.- United States launch vehicles.

7.1.3 Rocket Propulsion Theory

The earliest known written records of the basic laws of motion that were later applied to the field of rocket flight were discovered by the ancient Chinese. Widespread knowledge of these laws of motion did not exist in western civilization, however, until much later after the rediscovery of these laws by Isacc Newton in the 17th century. He formulated the law of gravity and the three laws of motion. These three laws of motion are defined as the following:

(1) If a body is at rest or moving at a constant speed in a straight line, it will remain at rest or keep moving at a constant speed unless acted on by a force.

(2) Force = (mass) (acceleration).

(3) For every action, there is an equal and opposite reaction.

Other than these basic laws of force, balance, and motion, three fundamental relationships of physics form the basis for understanding rocket propulsion theory. These laws of physics are:

(A) Conservation of mass
(B) Conservation of momentum
(C) Conservation of energy

Figure 7.1-6 below is used to used to illustrate these three relationships and derive the mathematical relationships that result from them in sections 7.1.3-2, 7.1.3-3, and 7.1.3-4.

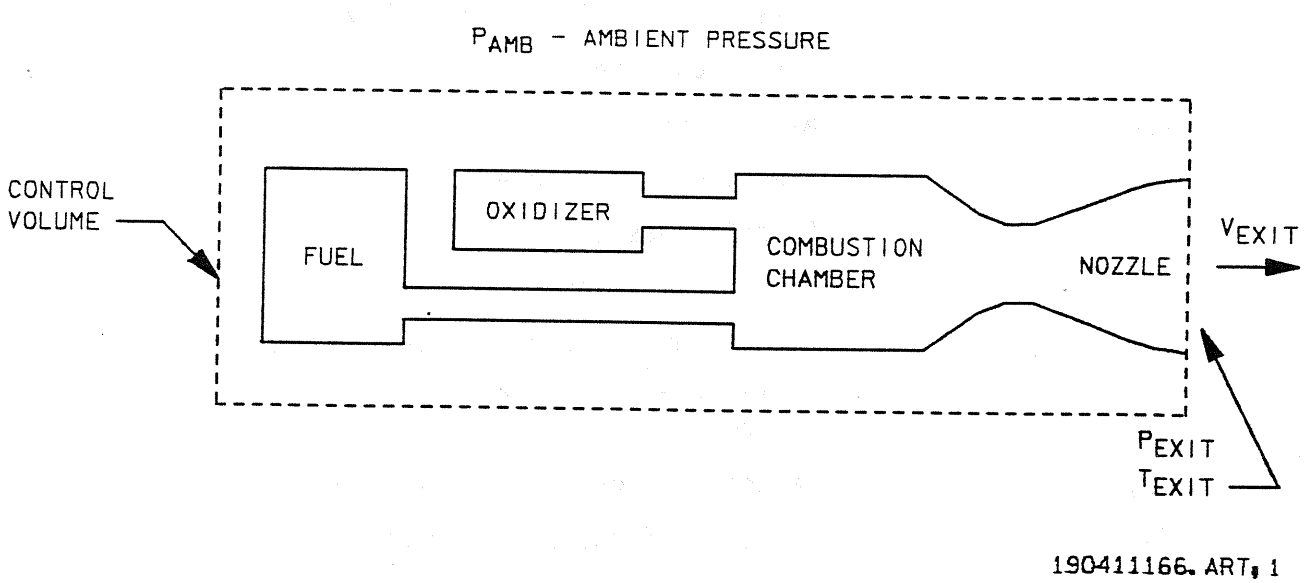


Figure 7.1-6.- Rocket propulsion system.

7.1.3.1 Symbology and Nomenclature

This section describes symbology used in the section 7.1.3

Symbol	Description
A_{exit}	Exit plane area of rocket nozzle
C_p	Specific heat at constant pressure
C_v	Specific heat at constant volume
$Enthalpy_{exit}$	Enthalpy of propellant combustion products at nozzle exit plane
$Enthalpy_{init}$	Enthalpy of propellant combustion products in nozzle combustion chamber
G	Acceleration of gravity in local environment
G_e	Earth standard acceleration of gravity (32.2 ft/sec-sec or 9.81 m/sec-sec)
$Height_{exit}$	Height of propellants as they leave the nozzle exit plane
$Height_{init}$	Height of propellants while they are in the combustion chamber
I_{sp}	Specific impulse of rocket motor
$\bar{M}$	Average molecular weight of propellant exhaust
$\dot{M}_{cv}$	Mass flow rate through control volume
$\dot{M}_{in}$	Mass flow rate into control volume
$\dot{M}_{exit}$	Mass flow rate out of control volume
P_{amb}	Ambient pressure outside control volume
P_{exit}	Pressure at nozzle exit plane
R	Specific gas constant
$\bar{R}$	Universal gas constant
T_{exit}	Static temperature of gases at nozzle exit
T_{init}	Static temperature of gases in combustion chamber
V_{exit}	Velocity of gases at nozzle exit plane
V_{init}	Velocity of gases in combustion chamber
$\dot{W}_{cv}$	Work done by control volume on outside system
γ	Ratio of specific heats for propellant gases

7.1.3.2 Conservation of Mass

The change of mass in any closed system, in this case the control volume in figure 7.1-6 which represents the rocket motor, can be expressed as the difference between mass inflow minus mass outflow. Expressed in mathematical terms:

(16.1) $$\dot{M}_{cv} = \dot{M}_{in} - \dot{M}_{exit}$$

For a pure rocket with all of its of the exhaust gases coming from onboard storage

(16.2) $$\dot{M}_{cv} = - \dot{M}_{exit}$$

7.1.3.3 Conservation of Momentum

The momentum equation can be derived by summing all the forces acting on the control volume. These forces include the momentum created by loss of mass from the control volume and the pressure differential across the control volume at the nozzle exit plane.

(16.3) $$\Sigma \text{ forces} = \text{Exhaust momentum thrust} + (A_{exit})\,(P_{exit} - P_{amb})$$

The exhaust momentum thrust in the change of momentum of the control volume caused by the change in velocity of the propellants as they leave the rocket nozzle exit plane.

(16.4) $$\text{Exhaust momentum thrust} = (\dot{M}_{exit})\,(V_{exit} - V_{init})$$

Since the inlet velocity with respect to the control volume is zero

(16.5) $$\text{Exhaust momentum thrust} = (\dot{M}_{exit})\,(V_{exit})$$

The force sum equation then translates to:

(16.6) $$\Sigma \text{ forces} = \text{Total thrust} = (M_{exit})\ (V_{exit}) + (A_{exit})\ (P_{exit} - P_{amb})$$

This term, the sum of the forces listed above, is referred to as the total thrust of a rocket. From equation 16.6, it can be seen that as a rocket gains in altitude, for a constant exit velocity and mass flow, the thrust increases as a result of decreasing ambient pressure.

A common term used to express rocket propellant performance is specific impulse or I_{sp}. Specific impulse is defined as:

(16.7A) $$I_{sp} = \left[V_{exit} + \frac{(P_{exit} - P_{amb})}{M_{exit}\ A_{exit}} \right] \Big/ G_e$$

For a perfectly expanded nozzle where $P_{exit} = P_{amb}$, this equation translates to:

(16.7B) $$I_{sp} = V_{exit}/G_e$$

Specific impulse defines the propellant performance per unit mass. For example, a specific impulse of 450 seconds, which the space shuttle main engines develop in a vacuum, means that each pound of propellant expelled can provide 450 lb of thrust for 1 second.

For single stage rockets, assuming no atmospheric drag and net gravitational acceleration, the change in velocity a rocket will have with a constant I_{sp} is defined by:

(16.7C) $$\text{Velocity change} = (I_{sp})\ (G_e)\ \text{LN} \left[\frac{(\text{Original Mass})}{(\text{Final Mass})} \right]$$

In equation 16.7C, the original mass is the total initial mass, including structure, payload, and propellant. The final mass is the mass at rocket burnout, including residual propellant, structure, and payload.

LN is the natural logarithm of the mass ratio.

See table 7.1-I for the effects of specific impulse and mass ratios on velocity change of single-stage vehicles.

TABLE 7.1-I.- ROCKET MASS RATIO VERSUS SPECIFIC IMPULSE

- Assumptions
  - No aerodynamic drag effects
  - Single stage vehicle
  - No net gravitational field effects
  - Constant specific impulse

Mass ratio Total initial mass / Total mass at burnout	Specific impulse (seconds)	Vehicle velocity change (ft/sec)
1.01	250	80
1.1	250	767
1.5	250	3260
2.0	250	5580
5.0	250	12960
10.0	250	18500
20.0	250	24120
30.0	250	27380
1.01	350	112
1.1	350	1070
1.5	350	4560
2.0	350	7810
5.0	350	18140
10.0	350	25900
20.0	350	33770
30.0	350	38330
1.01	450	144
1.1	450	1380
1.5	450	5870
2.0	450	10040
5.0	450	23330
10.0	450	33300
20.0	450	43420
30.0	450	49280

7.1.3.4 Conservation of Energy

The equations listed in the conservation of energy derivations below, 16.8 through 16.16, are made with the following assumptions:

- The working fluid behaves as a perfect gas.
- The chemical reactions take place under constant pressure.
- The gas expansion in the rocket nozzle is steady and isentropic.
- Molecular and ionic dissociation of combustion products is negligible. Chemical equilibrium exists throughout the system.

In the real world deviations from these assumptions can be significant, especially at extreme temperatures around 5000° F. The equations and relationships shown are still very useful, however, as they show the fundamentals of energy balance and exchange in a complex system.

The first law of thermodynamics applied to a control volume requires that the total energy transfer from a closed system be equal to the sum of the following:

- Chemical energy changes (e.g., combustion / enthalpy)
- Kinetic energy changes (e.g., gas velocity changes)
- Potential energy changes (e.g., change in height of fluid column)
- Work done by the system (e.g., turbopumps)

(Equation 16.8A)

$$\Sigma \text{ energy initial} - \Sigma \text{ energy exit} = 0$$

This equation expands into the following:

(Equation 16.8B)

$$\left[(\text{Enthalpy}_{init})(\dot{M}_{cv}) + (1/2)(\dot{M}_{cv})(V_{init})^2 + (\dot{M}_{cv})(G)(\text{Height}_{init})\right] -$$

$$\left[(\text{Enthalpy}_{exit})(\dot{M}_{cv}) + (1/2)(\dot{M}_{cv})(V_{exit})^2 + (\dot{M}_{cv})(G)(\text{Height}_{exit}) + \text{Work}_{cv}\right] = 0$$

The total energy change in a rocket motor between the combustion chamber and the nozzle exit plane must be zero based on the laws of conservation of energy (fig. 7.1-7). In most rocket engine systems, the potential energy change within the control volume (i.e., the (M) (G) (height) terms) is so small that the net change is negligible to the system change as a whole. Also, as the initial velocities of the propellants with respect to the control volume are zero and most of the work done to the system (frictional work and turbopump work) is returned back to the system in the form of heat or kinetic energy, equation 16.8B usually simplifies to:

(Equation 16.9)

$$(\text{Enthalpy}_{exit} + (V^2_{exit}/2)) = (\text{Enthalpy}_{init})$$

or

(Equation 16.10)

$$(V_{exit}) = \sqrt{(2.0)\ [(\text{Enthalpy}_{init}) - \text{Enthalpy}_{exit})]}$$

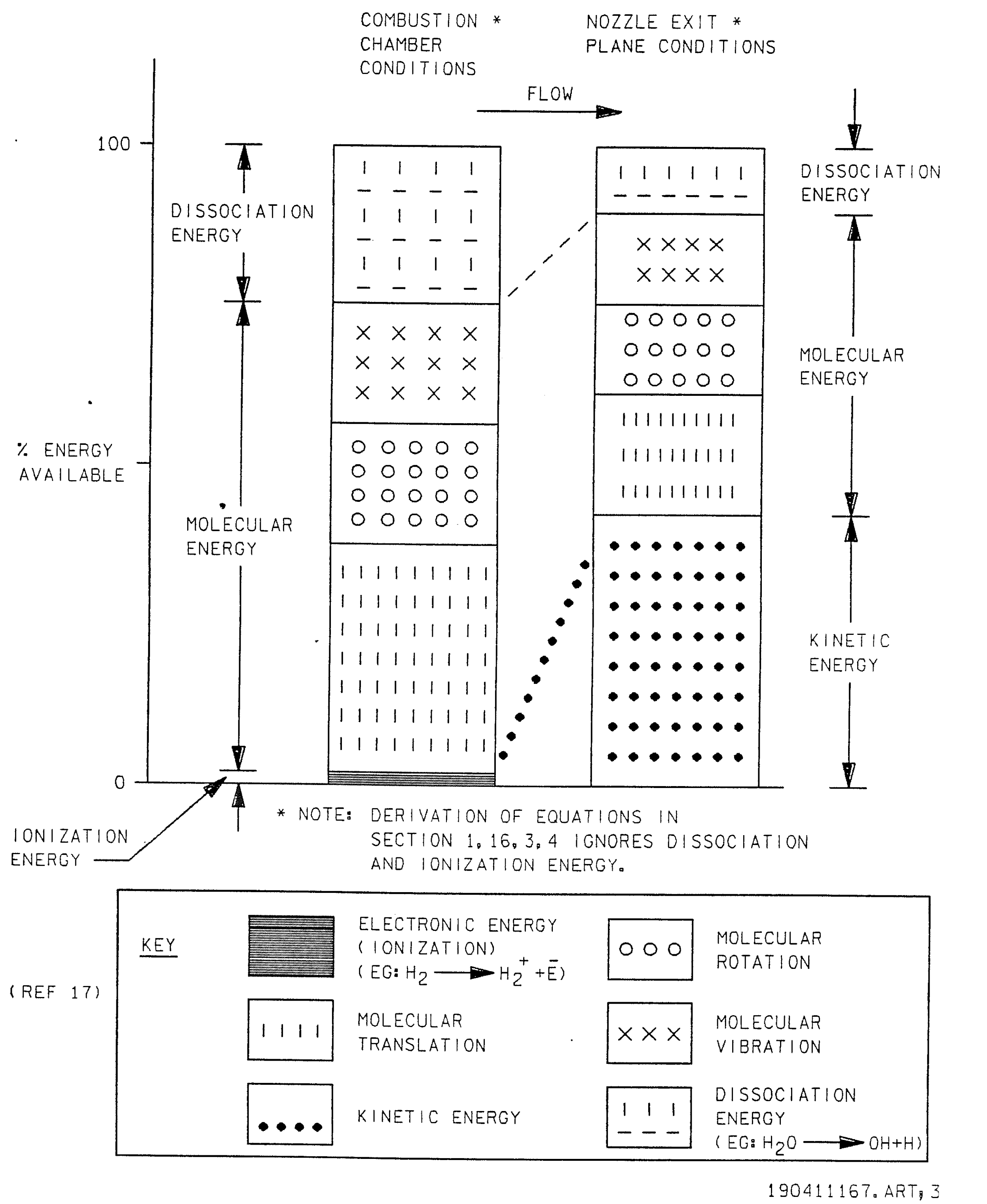


Figure 7.1-7.- Conversion of chemical energy to kinetic energy.

From equation 16.10 it can be seen that exit velocity is maximized by providing high initial enthalpy (high combustion temperatures and pressures) or extremely low exit enthalpy (low exit plane pressure and temperature). From the total thrust equation 16.6, it can be seen that for maximum thrust it is advantageous to have both high exit velocity and high exit plane static pressure. The energy equation, in 16.10, shows that these two terms always work in opposition to each other. That is, higher exit velocities take away energy that could be used in increasing exit plane pressure or conversely, allowing higher exit plane static pressures would reduce propellant exhaust velocity.

The highest rocket thrust is achieved by maximizing exhaust velocity and setting exit plane pressure as close to ambient pressure as possible, thus allowing most of the energy derived from propellant combustion to be converted into kinetic energy.

Using the relationship for enthalpy based on mass, specific heats at constant pressure, and temperature:

$$\text{(16.11)} \quad \text{Enthalpy} = (\text{Mass})\ (C_p)\ (T)$$

Applying equation 16.11 to equation 16.10 yields

$$\text{(16.12)} \quad V_{exit} = \sqrt{(2.0)\ (C_p)\ (T_{init} - T_{exit})}$$

For an ideal gas the gas constant is equal to the difference between the specific heat of constant pressure (C_p), minus the specific heat at constant volume (C_v)

$$\text{(16.13)} \quad R = \text{gas constant} = (C_p) - (C_v)$$

Using the universal gas constant $\bar{R}$, and molecular weight M

$$\text{(16.14)} \quad \bar{R} / M = (C_p) - (C_v)$$

Defining the ratio of the specific heats, γ, as $(C_p) / (C_v)$

$$\text{(16.15)} \quad \gamma = (C_p) / (C_v)$$

For a perfect diatomic gas (e.g., oxygen or nitrogen) $\gamma = 1.40$ at room temperature. In the case of a perfect monoatomic gas, (e.g., helium) $\gamma = 1.67$ at room temperature.

Using 16.14 and 16.15, equation 16.12 can now be converted to

$$(16.16)\quad V_{exit} = \sqrt{\left(\frac{2\gamma}{\gamma-1}\right)\left(R/M\right)\left(T_{init} - T_{exit}\right)}$$

From 16.16 it can be seen that the lower the molecular weight of the exhaust propellants, the higher the exhaust velocity will be and hence the greater amount of thrust per unit mass of the propellants.

7.1.4 Optimization of Liquid Rocket Engines

The choice of the proper design of a liquid rocket engine for a given mission is a very complex task. Quite often the requirements for reliability, cost, and performance will drive the design in conflicting directions. It is up to the engineer to weigh the benefits and costs of each design approach to determine the optimal design.

The purpose of this section is to serve as a guide to decide the right liquid rocket motor for a particular task. The guide covers five basic steps:

- 7.1.4.1 Is a Rocket Motor Needed At All?
- 7.1.4.2 Liquid Rocket Motors Versus Solid Rocket Motors
- 7.1.4.3 Optimization For Safety And Reliability
- 7.1.4.4 Optimization For Cost
- 7.1.4.5 Optimization For Performance

7.1.4.1 Step 1 - Is a Rocket Motor Needed At All?

Before beginning the rocket selection process, it is wise to determine if a rocket is needed at all for the task in question. Perhaps the tasks to accomplish could be handled better or with only a minor loss of performance by alternate means. For example, equipment packages on high altitude balloons or aircraft with high resolution may accomplish many of the objectives of sounding rockets, low-Earth-orbit missions, or distant planetary fly-by missions. Momentum wheels may substitute for pitch, roll, or yaw jets. Gravitational assists by other planets may reduce the need for high performance engine upper stages or high propellant loadings.

7.1.4.2 Step 2 - Liquid Rocket Motors Versus Solid Rocket Motors

There are unique advantages and disadvantages of using liquid rocket motors compared to solid rocket motors. The major distinctions are outlined below.

Advantages of liquid rocket motors:

1. Liquid rocket motors can be designed to throttle to different power settings on demand, although not all are made this way. Solid motors can be designed to a variable thrust level versus time profile, but it cannot be changed on demand as it depends on propellant casting shape.

2. Liquid motors can be designed to be restartable, although not all are made this way. Solid motors are not restartable.

3. Most liquid propellants have a higher specific impulse than solids.

4. Liquid propellants can be used to provide cooling jackets around the rocket motor nozzles and combustion chambers.

5. Solid motors require high strength case walls to contain high combustion pressures. This causes high structural weight. Pressure fed, but not turbopump fed, liquid rockets also have high structural weight.

6. Solid rockets motors that burn in low Earth orbit and generate nonvolatile combustion products (e.g., aluminum oxide) create long duration clouds of orbital particulate exhaust, which can damage other spacecraft. Some solid propellants also generate significant amounts of ozone depleting exhaust.

Disadvantages of liquid rocket motors:

1. Liquid rocket motors are more complex than solid motors with turbopump systems and/or gaseous tank pressure control systems.

2. Many liquid propellants are very corrosive and/or toxic.

3. Many liquid propellants, because of low boiling temperatures, have limited storage life under ambient temperature conditions.

4. Liquid propellants may require complex loading and conditioning systems if they are not storable.

5. Most liquid propellants have lower densities than solid propellants, requiring the design of rockets having more bulk volume and resulting in higher structural mass and greater aerodynamic drag.

An alternate concept is the hybrid rocket that combines some of the advantages and disadvantages of both these types of propulsion. It is a combination of liquid and solid propellant in the same rocket. This concept can use either a liquid fuel and solid oxidizer, or more commonly, a liquid oxidizer and solid fuel.

7.1.4.3 Step 3 - Optimization For Safety And Reliability

The first major area of optimization that should be looked at is the required safety and reliability of the rocket motor. A determination should be made of the needed reliability of the vehicle to have a viable mission and successful program. This should involve the following determination:

A. The required probability of completing the mission/program successfully

B. The required probability, if the mission/program fails, of recovering the crew or payload safely or affecting repairs on the payload by some alternate means

From a historical perspective, looking at United States, Soviet, and European launch systems used in putting payloads into Earth orbit or Earth escape, very few launch vehicles have had a loss rate in ascent less of than 1 in 25 attempts. None has had a loss rate less than 1 in 100. Table 7.1-II depicts the success of American launch vehicles between 1957 and 1990.

These high rocket loss rates should not be looked at as an insurmountable barrier, however. The parameters that influence these loss rates are under the control of the engineer. All rocket engine failures are the result of problems that fall into one or more of six basic problem areas. They are proper requirements definition, good design, comprehensive testing of all systems, manufacturing and production quality control and inspections, effective maintenance and updates, and proper operational techniques and use. These six areas apply to both software and hardware.

There is a hierarchical structure involved in these areas so that failures that exist in the first rungs, for example requirements definition, propagate through all subsequent areas (design through proper operational techniques). Likewise, failures in design do not determine requirements, but do carry through to adversely effect comprehensive testing through proper operational techniques. This hierarchical structure illustrates the important point that even a perfect design with complete redundancy cannot protect against failures that are the result of improperly defining the vehicle or mission requirements and understanding their environments. Likewise, a flawless manufactured piece of hardware that has a perfect record of maintenance can fail if the design or requirements were in error.

Therefore, when budgeting resources for engineering projects, the highest priority should be given to first establishing the right requirements and understanding the environments the vehicle will be experiencing. If the right questions are not properly defined, the right answers cannot be found.

TABLE 7.1-II.- RELIABILITY OF UNITED STATES LAUNCH VEHICLES*

Year	Launch attempts to orbit or Earth escape	No. of successful launches	Percent successful
1957	1	0	0
1958	17	7	41
1959	19	11	58
1960	29	16	55
1961	41	29	71
1962	59	52	88
1963	46	38	83
1964	64	57	89
1965	70	63	90
1966	77	73	95
1967	61	58	95
1968	48	45	94
1970	30	29	97
1971	35	32	91
1972	33	31	94
1973	25	23	92
1974	25	24	96
1975	31	28	90
1976	26	26	100
1977	26	24	96
1978	33	32	97
1979	16	16	100
1980	15	13	87
1981	19	18	95
1982	18	18	100
1983	22	22	100
1984	22	22	100
1985	18	17	94
1986	9	6	67
1987	9	8	89
1988	12	12	100
1989	18	18	100
1990	27	27	100
Total	1042	935	89.7 %

A breakdown by decade illustrates the following:

1957 - 1959	37	18	48.6 (1 : 1.9 loss)
1960 - 1969	535	471	87.9 (1 : 8.3 loss)
1970 - 1979	280	265	94.6 (1 : 18.5 loss)
1980 - 1989	162	154	95.0 (1 : 20.0 loss)

* All known United States launch attempts to orbit from both liquid and solid propellant motors are included: NASA/DOD/commercial/scientific. Both manned and unmanned missions are included. Success is defined as delivery of payload(s) into orbit or Earth escape, while payloads destined for suborbital missions are not included.

If designing a liquid rocket motor and propellant system optimized for safety and reliability the following selection guidelines should be used:

1. Understand well the environment in which your system will operate.

   This includes the ambient environment as well as the thermal, stress, aerodynamic, shock and vibration, pressure, flow loadings, and other system parameters on your motor system. The motor should be tested over the full range of environmental extremes it will see in both nominal and off nominal conditions.

2. Provide independent teams to determine mission/vehicle requirements and environmental conditions. Redundancy should apply to development of not just hardware and design but also requirements development tasks.

3. Keep the design simple.

   The simpler the system, the less likely people are to get confused and make mistakes with the hardware, software, and operations. Also, the simpler the system, the more likely major issues are to be seen and resolved in the early, preliminary design stage and not be hidden or misunderstood until the critical design stage is reached. A major issue is that in an aerospace engineering environment, where typically 10 to 15 percent annual workforce turnover exists, it is difficult to get an experienced work team to deal with extremely complex systems. If it takes 3 or 4 years experience to gain a good understanding of a system, and 30 to 60 percent of the workforce leaves in this same duration, it is playing a hopeless game of catchup.

   Taking one example of a design selection, liquid rockets using pressure-fed systems through series-parallel valves are more reliable and simpler than complex turbopump-fed engine designs.

4. Use adequate safety margins.

   The higher the design safety margins, the more reliable the vehicle will be. Most present day designs apply safety factors of about 1.4 to 1.5 for ultimate stress loads (based on worst case loads) for most major structural components. Using safety factors of 1.7 or even 2.0 could enhance safety, but at the expense of considerable weight. Safety factors also could be enhanced for fatigue, corrosion resistance, ablation wear, vibration loads, and thermal protection.

5. Develop redundant hardware systems, perhaps even multiple backup systems, in case primary systems fail. However, guideline 5. is heavily dependent on guideline 4. If the primary failure modes of your systems are catastrophic, uncontained damage rather than benign shutdown, having system redundancy can actually increase your risk of failure rather than improve it. For example, if a turbopump is more likely to explode rather than shut down safely with the most likely failure scenario, having four turbopumps running (of which only three are required for a safe flight) during a nominal mission would be worse

than with three turbopumps and no backup systems. Redundancy should apply to software as well as hardware.

6. Adopt a philosophy that states that systems are unsafe and unreliable until proven otherwise.

   Do not assume a system is safe if no indications of major problem areas exist, rather assume a system is unsafe until data can be found to prove a system is reliable. Absence of data on a system should not be interpreted as meaning everything is designed properly or operating well.

7. Install escape systems/abort scenarios for crew and payload.

   High reliability systems should be developed to return crew and payload safely in the event of major malfunctions of the vehicle.

8. Quality control and nondestructive testing.

   By subjecting flight hardware to extensive testing before flight, most types of flaws and maintenance errors can be discovered. Systems tests should be carried out on entire integrated hardware with flight software and flight loads. This will protect against flaws that exist in subsystem interface areas that could remain hidden if only subsystem components are tested individually. Extensive inspection techniques exist, such as X-ray scans, ultrasonics, thermography, laser scans, dye penetrates, pull tests, nuclear magnetic resonance, and fiber optic boroscopes.

   Quality control should be given a very high priority with properly trained and certified personnel. Clean rooms and high quality contamination control systems should be used on sensitive components.

9. Keep good communications channels open.

   All personnel should have responsive and effective means to suggest changes, report problems, and gain easy access to information needed to carry out their job responsibilities. Systems should be in place so information flows rapidly to the proper personnel rather than running through a long series of in and out boxes lasting weeks or months.

10. Ensure that rocket stages/engines and propellant exhaust do not cause long term debris hazards in Earth orbit that will threaten the safety and reliability of other vehicles.

11. Use propellants that are the least hazardous and toxic to ground and flight personnel.

7.1.4.4 Step 4 - Optimization For Cost

Many design and program strategies can be used to bring about lower rocket engine costs. Among them are:

1. As in Step 3, keep the design as simple as possible.

2. An extremely high percentage of the cost of any engine is a result of research and development costs. When possible, using rocket motors already designed and built rather than developing new ones will save considerable money.

3. If multiple stages are used, every effort should be made to use similar engine systems on the different stages to save costs.

4. Design systems with future growth and improvement in mind. This should involve interchangeability of parts, ease of maintenance and checkout of engine systems.

5. Use only hardware that is available from multiple vendors. Single source suppliers of critical hardware leave any program too vulnerable.

6. Production of large lots of motors allows research and development costs to be spread out among many units, allowing much lower costs on a per unit basis.

7. For ground based launch systems, finding a launch site close to the equator (except for polar orbit missions) will greatly reduce propellant/rocket liftoff mass.

8. High altitude launch sites, balloons, and aircraft launches can sometimes save on costs by lowering the cost in propellant rockets use in fighting air drag and offering higher thrust through lower back pressure. This must be balanced against the cost of developing and maintaining a high altitude launch site/platform.

9. Determine the tradeoffs between reusable and expendable launch systems. Reusable vehicles can greatly reduce manufacturing costs for hardware, but quite often at the expense of much greater system refurbishment and maintenance costs. For example, it costs more to refurbish a household lightbulb than buy a new one.

7.1.4.5 Step 5 - Optimization For Performance

The following guidelines can be used to maximize the performance of rocket motors. Maximization of performance is interpreted to mean designing a rocket to get the most thrust per unit mass of propellant and obtaining a rocket that will need the minimum amount of propellant to achieve its required terminal velocity. Quite often, guidelines to improve performance will conflict with those that offer greater reliability and cost savings.

1. Higher combustion temperatures and pressures will allow greater propellant exhaust velocities.

2. The lower the molecular weight of the exhaust gases, the greater the propellant exhaust velocities.

3. The lower the ambient pressure and the more closely the exhaust nozzle exit pressure matches ambient pressure, the higher the rocket engine thrust per unit mass will exist. See figure 7.1-8.

4. Except for liquid motors having short thrust durations, turbopump-fed propellant systems will weigh less than pressure-fed propellant systems.

5. The higher the operating pressures of the combustion chambers and turbopumps, the lower the weight of the rocket motor for a given thrust.

6. Multistage combustion offers higher performances than single-stage (gas generator) combustion systems.

7. For Earth-launched systems, there is an optimal rocket burn time for best performance. A minimum amount of propellant is expended fighting gravity by having as rapid an acceleration (i.e., shortest burn time) as possible. However, several factors influencing performance are favored by long burn times. They include less structural mass to withstand lower "g" forces, lower aerodynamic and thermal loads in ascent, and smaller propellant flow rates, resulting in lower weight gaseous pressure control systems or turbopumps.

8. Depending on mission, multiple stages may offer considerable weight and propellant savings over single stage vehicles, especially where terminal velocities are large.

9. The lower the structural mass of the rocket, the greater the performance.

10. Higher density propellants will reduce storage volume, and hence lead to less structural mass requirements and less aerodynamic drag.

11. For launches into equatorial orbits, launches at as low a latitude as possible will greatly save on propellant.

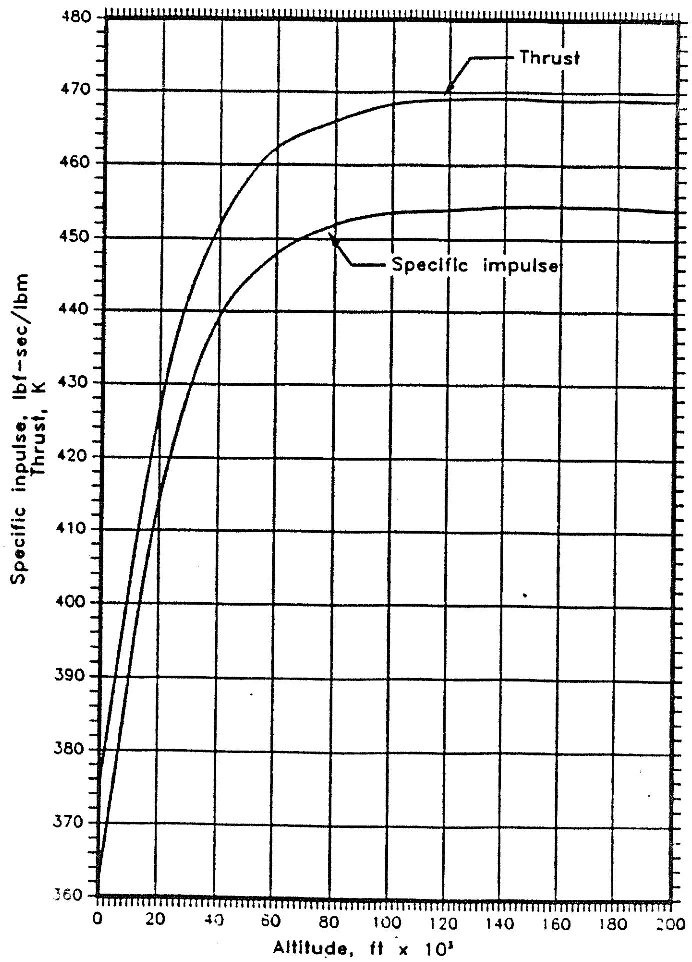


Figure 7.1-8.- Nominal specific impulse and thrust as a function of altitude. Space shuttle main engine 100 percent power level.

7.1.5 Liquid Rocket Motor Hardware

The next several sections cover the basic hardware that liquid rockets use. Different design concepts are discussed on propellants, injectors, turbopumps, pressure-fed systems, combustion chambers, structural cooling techniques, rocket nozzles, and thrust vector control systems. Table 7.1-III summarizes the major design options in liquid rockets.

7.1.5.1 Liquid Propellants

Chemical energy needed to accelerate the rocket exhaust to high velocities comes directly from the chemical reactions of the propellants in the combustion chamber. Liquid rockets derive their energy from either self-combusting compounds, called monopropellants (e.g., nitromethane), bipropellants containing a fuel and oxidizer, or even tripropellants in some cases.

Some rocket engines also work in a mixed-mode combustion, allowing one type of fuel and oxidizer during the first part of operation, followed by another fuel and oxidizer over the latter part of the motor burn. This allows a particular fuel/oxidizer to be chosen to optimize each flight regime rocket is operating.

Propellant decomposition or combustion can be initiated by either contact with a suitable catalyst, exposure to a spark or heat source, or, in the case of hypergolic propellants that ignite upon contact with each other, by good mixing conditions.

Qualities that are desirable for propellants include:

- High combustion temperatures and pressures
- Low molecular weight of exhaust propellants
- Fluid easily atomized/vaporized in combustion chamber
- Fast combustion (rapid flame) velocities
- High storage densities to minimize tank size and aero drag
- Good long term storability, depending on mission
- Low toxic and corrosive effects
- Minimal damage to the local and global environment from exhaust gases and propellant spills
- Large quantity available from multiple sources
- Ease of handling by ground support facilities
- Low cost for both production and storage
- For combustion chambers or rocket nozzles using liquid propellant for a coolant, propellants having a high heat capacity and high heat transfer rate are best. Generally fuels are far better, and safer, coolants than oxidizers
- Chemically stable, resistant to explosion by shock or heat

Tables 7.1-IVa and 7.1-IVb present a list of many common propellants with specific characteristics on each.

TABLE 7.1-III.-DESIGN OPTIONS IN LIQUID ROCKETS

Rocket system	Section	Major design options
Propellants (Monopropellants) self combusting	(1.16.5.1)	Hydrazine - three types Nitromethane Hydrogen peroxide Ethylene oxide
(Bipropellants) Commonly used fuels	(1.16.5.1)	Liquid hydrogen Hydrazine - three types Kerosene (RP-1) Ethyl alcohol Ammonia Liquid methane
Commonly used oxidizers		Liquid oxygen Nitrogen tetroxide Hydrogen peroxide Nitric acid - two types
Injectors	(1.16.5.2)	Unlike stream impingement Like stream impingement Shower Swirl
Propellant flow systems	(1.16.5.3)	Pressure fed Turbopump Electric pump
Combustion and structural cooling systems	(1.16.5.4)	Single/multistage combustion Equilibrium & frozen flow Refractory wall Regenerative cooling Transpiration cooling Ablation cooling Film cooling Heat sink Radiation cooling
Control valves	(1.16.5.5)	Electromechanical Pneumatic Hydraulic Action envelopes Thermoreactive Mechanical Pyrotechnic
Nozzles	(1.16.5.6)	Standard bell or cone Plug or aerospike nozzle Expansion deflection Variable expansion ratio
Thrust vector control	(1.16.5.7)	Control fins in airstream Nozzle gimbaling Secondary flow injection Exhaust gas control vane Differential engine thrust

TABLE 7.1-IVa.- LIQUID ROCKET PROPELLANT CHARACTERISTICS

FUELS

Compound	Notes*	Anhydrous hydrazine	Monomethyl-hydrazine (MMH)	Unsymmetrical dimethyl-hydrazine (UDMH)	Nitromethane	Ethylene oxide	Liquid hydrogen	Kerosene (RP-1)	Ethyl alcohol (100%)
Liquid color		Clear	Clear	Clear	Clear	Clear	Clear	Clear	Clear
Odor		Ammoniacal	Ammoniacal	Ammoniacal	—	Pleasant	None	Petroleum	Ethereal
Formula		N_2H_4	CH_3NHNH_2	$(CH_3)_2NNH_2$	CH_3NO_2	$(CH_2)_2O$	H_2	$CH_{(1.95-2.0)}$	C_2H_5OH
Molecular weight		32.05	46.08	60.10	61.04	44.05	2.02	App. 175	46.07
Density (grams/cc)	(a)	1.01	0.877	0.793	1.13	0.870	.071	.806	.789
Freezing point (F/C)	(b)	34.8/1.6	-62.0/-52	-71/-57	-20.2/-29	-167/-111	-435/-259	-48/-44	-174/-114
Boiling point (F/C)	(c)	236/113	190/89	146/63	214/101	51.3/10.7	-423/-253	400/204	173/78
Autoignition temp. (F/C)	(d)	518/270	382/194	480/249	500/260	1058/570	N/A	N/A	N/A
Heat of vapori-zation (Btu/lb)	(e)	602	377	251	—	—	192	106	360
Specific heat capacitY (Btu/lb/f)	(f)	0.74	.70	.65	—	—	2.34	.46	.578
Vapor pressure (psia @ 68° F)	(g)	0.20	0.70	2.38	N/A	N/A	N/A	.02	.85
MAC (ppm)	(h)	1.0	<0.50	0.50	100	100	—	—	—
Adiabatic flame temperature (F) and ISP (sec) as monopropellant	(i)	1097° F 186 sec	—	—	4002° F 244 sec	1760° F 189 sec	N/A	N/A	N/A
Birpropellant with labeled compound with : O/F ratio : temp (F) : ISP (sec) :	(j)	LO_2 0.75 5370 265	—	LO_2 1.4 5650 249	—	—	LO_2 8.0 5870 360	LO_2 2.35 5200 240	—

*See notes on page 7.1-33.

TABLE 7.1-IVb.- LIQUID ROCKET PROPELLANT CHARACTERISTICS

		Fuels		Oxidizers				
Compound	Notes*	Ammonia	Methane	Liquid oxygen	Nitrogen tetroxide	Hydrogen peroxide (100%)	Red fuming nitric acid (RFNA)**	White fuming nitric acid (WFNA)**
Liquid color		Clear	Clear	Light blue	Yellow	Clear	Brown	White
Odor		Pungent	None	None	Noxious	None	Pungent	None
Formula		NH_3	CH_4	O_2	N_2O_4	H_2O_2	82-85% HNO_3 13-15% N_2O_4 2-3% H_2O	97.5% HNO_3 0-0.5% N_2O_4 2% H_2O
Molecular weight		17.03	16.03	32.0	92.02	34.02		
Density (grams/cc)	(a)	0.604	0.415	1.15	1.45	1.45	1.548-1.551	1.468-1.558
Freezing point (F/C)**	(b)	-108/-78	-297/-183	-362/-219	11.8/-11.2	31.3/-0.4	-76 to -61 (°F)	-44 to -43 (°F)
Boiling point (F/C)**	(c)	-28/-33	-263/-164	-298/-183	70.1/21.2	302/150	140 (°F)	191 (°F)
Autoignition temp. (F/C)	(d)	1204/651	N/A	N/A	N/A	—	N/A	N/A
Heat of vaporization (Btu/lb)	(e)	590	219	92	178	653	—	270
specific heat capacity (Btu/lb/f)	(f)	1.125	0.825	0.406	0.368	0.631	0.415	0.423
Vapor pressure (psia @68° F)	(g)	N/A	N/A	N/A	14.0	<0.10	2.57	0.924
MAC (ppm)	(h)	100	20,000		5.0	1.0	10	10
Adiabatic flame temperature (F) and ISP (sec) monopropellant	(i)	N/A	N/A	N/A	N/A	1839° F 165 sec	N/A	N/A
Bipropellant with labeled compound with : O/F ratio: temp (F) : ISP (sec):	(j)	LO_2 1.3 4940 255		Kerosene 2.35 5200 240	N_2H_4 1.2 5000 250	N_2H_4 1.7 4690 255	N_2H_4 1.3 4980 247	UDMH 2.7 5100 240

* See notes on page 7.1-33.
** RFNA and WFNA freezing and boiling points in °F only. F/C = Fahrenheit/centigrade.

Notes for tables 7.1-IVa AND 7.1-IVb

(a) Density reflects conditions at the local boiling temperature at 1 atmosphere pressure for cryogenic propellants. For propellants that are liquids at 1 atmosphere at 68° F, the density at this point is given instead.

(b) Freezing conditions are at 1 atmosphere pressure.

(c) Boiling conditions are at 1 atmosphere pressure.

(d) Autoignition temperature is the temperature at 1 atmosphere pressure at which uncontrolled decomposition and/or explosion occur.

(e) Heat of vaporization is the amount of Btu's needed to transform 1 pound of liquid propellant to the gaseous phase at 1 atmosphere pressure at either the local boiling temperature (ammonia, methane, LO2, N_2O_4, LH_2 RP-1, ethyl alcohol) or from 77° F (H_2O_2, WFNA, all hydrazines).

(f) Specific heat capacity is the amount of Btu's needed to raise 1 pound of liquid propellant 1° F. The temperature state it applies is either at the boiling point for cryogenic propellants, or 68° F for non-cryogenic propellants.

(g) Vapor pressure is the equilibrium pressure that would exist in a closed container with nothing but the propellant in existence at 68° F. For cryogenic propellants, which would vaporize completely at room temperatures, the partial pressure would exceed 14.7 psia, so these values are not given.

(h) MAC is the maximum acceptable concentration of propellant, in parts per million (ppm), that would be suitable for humans to be in continuous exposure without toxic effects.

(i) & (j) Flame temperatures and specific impulse values shown assume combustion chamber pressures of 1000 psia and expansion to 14.7 psia in the nozzle exit plane. Ambient pressure is assumed to be 14.7 also, so no overexpansion or underexpansion is assumed to exist. O/F ratio is the oxidizer to fuel ratio by weight.

7.1.5.2 Injectors

Injectors are fine nozzles that spray propellants into the combustion chamber. They are best designed to be short tubes with rounded inlets to keep flow turbulence and friction drop minimal. They should provide a very uniform mixture of propellants in the chamber and vaporize the propellants sufficiently for quick combustion. Often, by careful design, it is possible to provide a protective circumferential layer of lower temperature gas near the walls of the combustion chamber that prevents excessive convective and conductive heat transfer into the structural walls.

The four basic types of injector designs (fig. 7.1-9) are:

- Like stream impingement
- Unlike stream impingement
- Shower head nozzle
- Swirl nozzle

All four types have been used successfully in rocket engines. Unlike stream impingement works well with hypergolic propellants to provide a quick ignition front. Swirl injectors work well with monopropellants to provide adequate fluid vapor distribution as no mixing with other fluid streams is required. Occasionally, fluid streams will be focused on short metal stubs called splash plates to enhance mixing and atomization of propellants.

7.1.5.3 Propellant Flow Systems - Pressure-Fed And Pump-Driven

In order to obtain high thrust and high propellant exhaust velocities, it is necessary to have the propellant chemical reactions occur at high pressure in the combustion chamber. This requires the propellants to be supplied by a system that imparts a high pressure to them through the injector face. The pressure provided to the propellants must be high enough that pressure rise and oscillations in the combustion chamber do not interfere with the propellant flow rate into the combustion chamber. If a coupling between the combustion process and feed system occurs through pressure oscillations, a low frequency (less than 100 cycles per second) instability will occur. This low frequency pressure and thrust oscillation is called chugging. It can be corrected by a higher injector face pressure drop (ref. 17).

The required injector drop can be provided by either a pressure-fed system or propellant pumps. Pressure-fed systems are far less complex, but for long duration burns or large propellant volume, pumps can usually provide rocket structures that are much lighter in weight and higher in performance. See table 7.1-V for comparison of pressure-fed and turbopump systems.

In cases where low propellant flow and small pressure head rise is required, electric driven pumps can be used rather than turbopumps. They can be made quite reliable and efficient for low flow systems. This must be balanced against the ability of the rockets to produce electrical power.

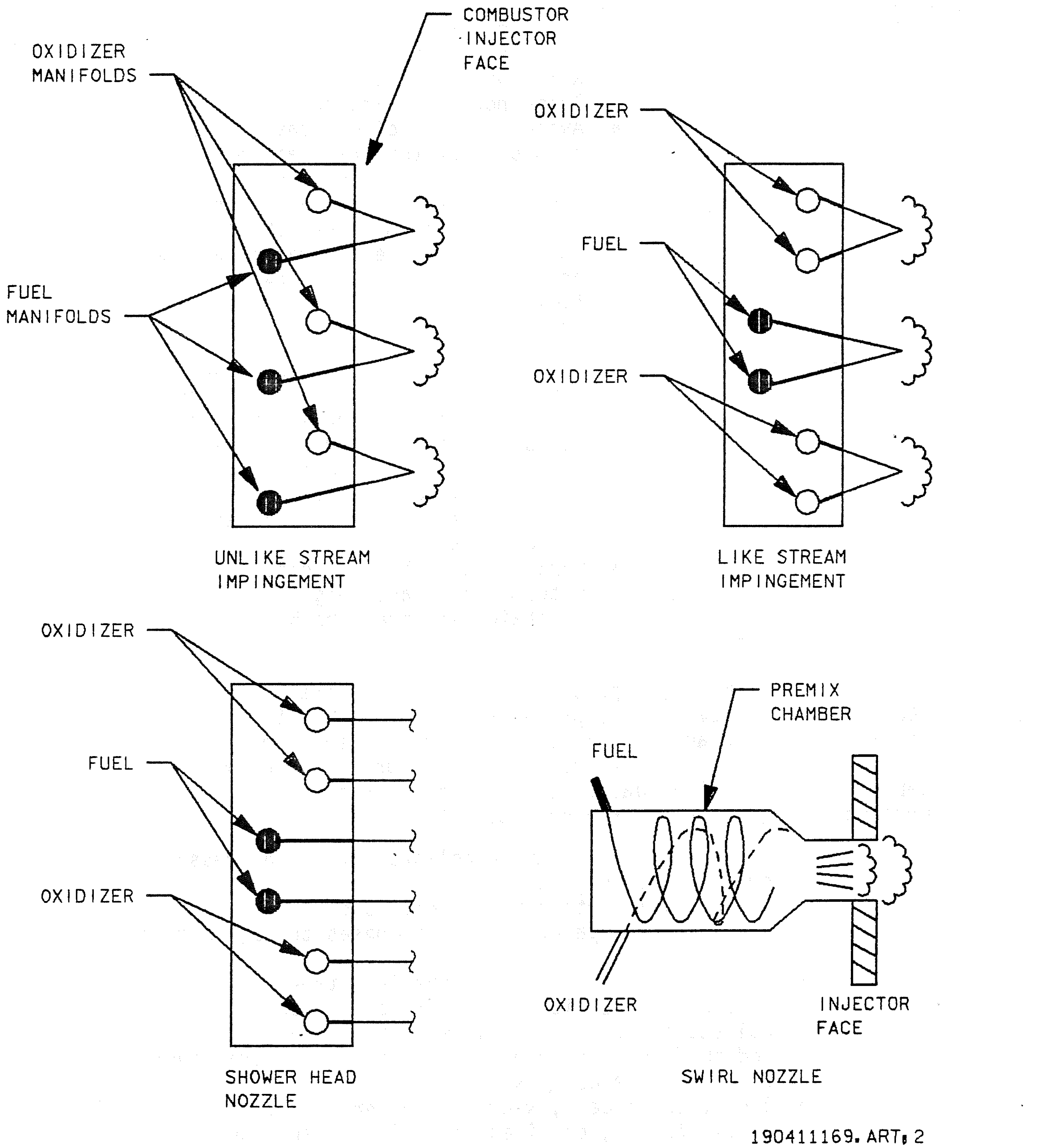


Figure 7.1-9.- Injector designs.

TABLE 7.1-V.- PRESSURE-FED AND TURBOPUMP SYSTEMS

System	Advantages
Pressure-fed systems	• Low complexity • High reliability • Works well at low ambient pressures with large nozzle expansion ratios • Avoids problem of cavitation that turbopump systems have at low NPSP (net positive suction pressure) • Works well with high density propellants (less tank volume) over short time periods • As turbopump size and weight are dependent on maximum propellant flow rate, higher flow needs (i.e., high thrust) make pressure-fed systems more attractive.
Turbopump systems	• Lightweight propellant tanks save on vehicle structural weight. Thinner walled tanks needed to maintain tank pressure • Works well at high ambient pressure • Works well with low density propellants over long thrust durations • Requires less pressurizing gas to provide adequate tank pressure • Works best with lower propellant flow rates as size and weight of system drop to maintain same pressure head

Propellant systems that operate under high acceleration can easily separate fluid from gaseous propellant to operate. Systems which operate under low acceleration, or provide attitude control, or are just starting from a zero-g condition have unique problems in separating liquid from gas to start. When working in a low or zero-g environment, options to provide liquid flow include:

- Fluid filled bladder or bellows with high pressure gas envelope to squeeze liquids out
- Centrifugal suction pump, which picks up both gas and liquid, and separates the two phases through high speed centrifugal force
- Spinning the vehicle or propellant tanks to draw the liquids to the side walls for collection
- Collection screens and lines, which use viscous effects and surface tension effects of fluids to draw them out from the gaseous phase
- Bootstrap process, where a low thrust system, solid or liquid fueled, provides initial acceleration to separate fluid and gas in a large propellant tank that will subsequently be used once settling is complete

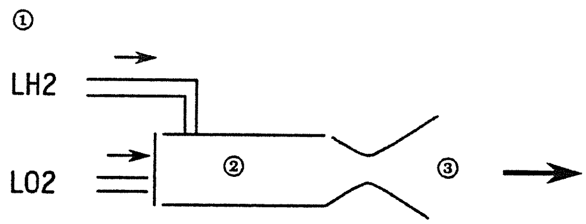
High frequency acoustic instabilities are a major problem in rocket engine combustion chamber and nozzle design. These frequencies typically are more than 1000 cycles per second and are called screaming. They are dependent on injector spray patterns, combustion mode, and chamber and nozzle geometry. Quite often baffles or injector flow patterns are used to dampen out the probability of propellant screaming.

The exhaust gases that are generated in the combustion chamber rarely are in chemical equilibrium as a result of the extreme temperatures, pressures, and short residence time before exiting the rocket. Almost always, molecular disassociation, and sometimes ionic disassociation, occurs to a high degree in the exhaust gas products.

When the percentage of each molecular and ionic species in the nozzle exit plane exhaust equals that of the combustion chamber a state known as frozen flow exists (fig. 7.1-10).

When recombination of the molecular and ionic species occurs to a high degree between the combustion chamber and nozzle exit plane, such that the molecular and ionic makeup at the exit plane is in equilibrium with the exit plane temperature and pressure, a state of equilibrium flow exists (fig. 7.1-10).

Typically, in most rockets, an intermediate state between frozen flow and equilibrium flow exists. This state is called the nonequilibrium flow condition. Equilibrium flow allows the most specific impulse to be obtained from the propellant, as more combustion energy is converted into kinetic energy and less wasted on increasing molecular and ionic dissociation. Long exhaust nozzles can enhance equilibrium flow conditions as they allow greater residence time for recombination, but at the expense of more boundary layer friction losses and higher nozzle weight.



Typical molecular composition

Numbers listed are in molar fractions for both reactants and products.

Location	H_2	O_2	O	H	OH	H_2O
① Inlet	2.0	1.0	–	–	–	–
② Combustion (6300° F/300 psia)	.32	.10	.05	.11	.23	1.51
③ Nozzle exit (frozen flow)	.32	.10	.05	.11	.23	1.51
③ Nozzle exit (equilibrium flow)	.10	.03	.01	.02	.08	1.85

Figure 1.16-10.- Frozen and equilibrium flow.

7.1.5.3.1 Pressure-fed systems. - Pressure-fed propellant systems are the simplest to build and operate for rockets. They provide a high pressure gas head over the liquid propellants great enough to drive them through the injector and into the combustion chamber. Because the tank pressure must be higher than the combustion chamber pressure, strong, heavy, propellant tanks are required. The high pressure gas must not react with the propellant so either a relatively inert gas or propellant (fuel or oxidizer) bled off from the combustion process or heat exchanger is used as the pressurizing medium. Sometimes an inert gas is used for initial tank pressure control followed by gaseous propellant to provide a pressure head during the majority of the engine burn.

Usually the inert gases of choice are either nitrogen or helium. As nitrogen liquefies at -321° F (at 1 atmosphere), it cannot be used as a pressurant for cryogenic propellants near this temperature. Helium 4, the most common helium isotope, liquefies at -452.4° F (at 1 atmosphere) and can be used with any oxidizer or fuel. Inert gases are generally stored in high pressure tanks at 2000 to 5000 psia.

When propellants are used as pressurants, they are generally obtained from fuel or oxidizer bled off or routed through turbopumps, heat exchangers, or nozzle and combustion chamber cooling channels. Sometimes onboard gas generators can be used where combustion products in a gaseous phase are routed back to the propellant tanks. Extreme care must be used with gaseous oxidizers as many of them are extremely reactive with many metals and polymers that are normally considered inert.

The tank pressure is maintained by controlling the pressurant addition by using flow control valves, regulators, or orificed channels. Relief valves are used to protect against tank overpressure from regulator(s) or flow control valves(s) failed open. Check valves are placed to protect against failures that result in hazardous backflow conditions. In-line filters are inserted to block out any contaminants. Test ports are designed in at several locations to verify the system integrity preflight and perform leak checks.

7.1.5.3.2 Turbopump systems.- A good summary on the basics of turbomachinery is provided in systems brief 1.4.

High pressure injection of both liquid oxidizers and fuels can be provided very efficiently by turbopump systems. Turbopumps work by providing a large pressure increase to the propellant above the nominal propellant tank ullage pressure by running the fuel or oxidizer through single or multiple stage compressor blades. These compressor blades are coupled to a shaft that is connected to a high speed turbine.

A system for generating high pressure gas to spin the turbine is used to power the fuel and oxidizer turbopumps. This hot gas can come either from combustion of the primary propellants used to provide the rocket thrust or a separate propellant supply system. The turbine exhaust gas can either be ejected as a separate stream or intermixed with the propellant exhaust and sent through the primary rocket nozzle. Both axial and centrifugal pumps have been used in rocket turbopump systems (fig. 7.1-11).

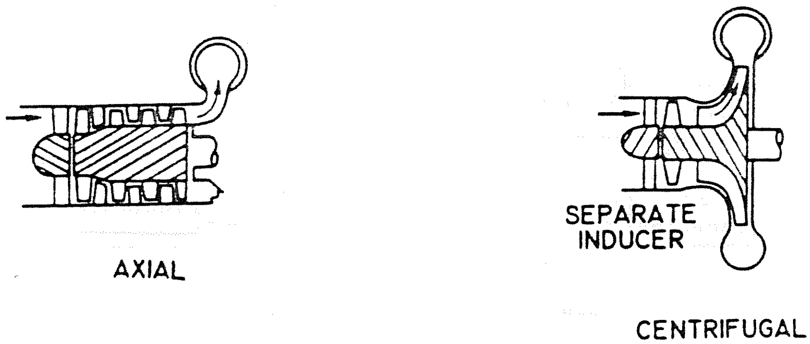
With low pressure rise rates required, going from tank ullage gas pressure to combustion chamber pressure levels, a single fuel pump and oxidizer pump can be used. For extremely high combustion chamber pressures, pumping requirements can best be met by having dual turbopumps, low and high pressure, in both the fuel and oxidizer systems, for a total of four turbopumps per engine. In this configuration, the output of the low pressure pump serves as inlet to the high pressure pump (fig. 7.1-12).

While greatly reducing rocket stage weight in many cases, turbopumps also produce a new set of concerns that are not found in pressure-fed propellant systems. These include:

- Turbine blade or compressor blade failure from excessive stress, fatigue, or high temperature
- Cavitation resulting in uncontained damage in many cases if early propellant depletion occurs or net positive suction pressure becomes too low (systems brief 1.4)
- Bearing failure
- Static and dynamic seals which are extremely complex

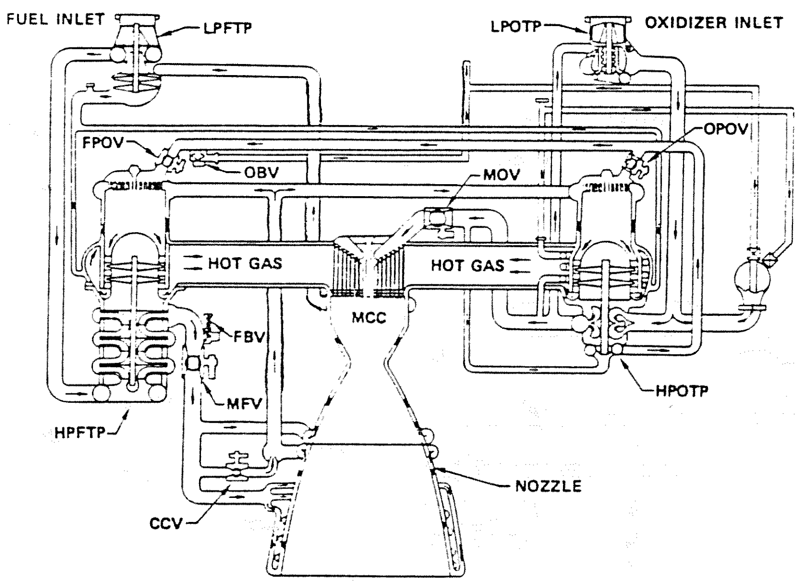
7.1.5.4 Combustion And Structural Cooling Techniques

Combustion processes for rocket engines can occur either solely in a main combustion chamber or through the use of a staged combustion cycle. In staged combustion systems, propellants undergo partial burning in one area, are sent through turbines, and then sent into another chamber for more complete combustion (figs. 7.1-13a and 7.1-13b). Staged combustion offers more efficient propellant use through better combustion and higher pump pressures than single stage systems (fig. 7.1-14). However, this comes at the expense of higher cost, increased complexity, and higher probability of combustion and acoustic instabilities.



(ref. 18)

Figure 7.1-11.- Axial and centrifugal turbopump designs.



LPFTP - low pressure fuel turbopump
LPOTP - low pressure oxidizer turbopump
MCC - main combustion chamber
MFV - main fuel valve
MOV - main oxidizer value
OBV - oxidizer bleed valve
OPOV - oxidizer preburner oxidizer valve
CCV - chamber coolant valve
FPBV - fuel preburner valve
FPOV - fuel preburner oxidizer valve
HPFTP - high pressure fuel turbopump
HPOTP - high pressure oxidizer turbopump

Figure 7.1-12.- Space shuttle main engine.

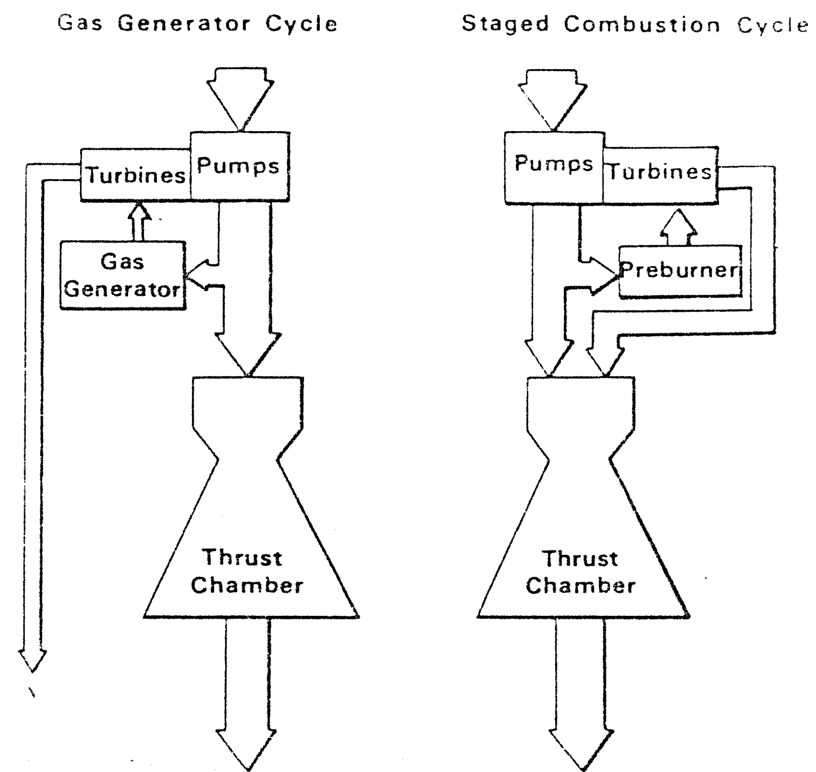


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Figure 7.1-13a.- Gas generator (single-stage combustion) versus preburner (multistage combustion).

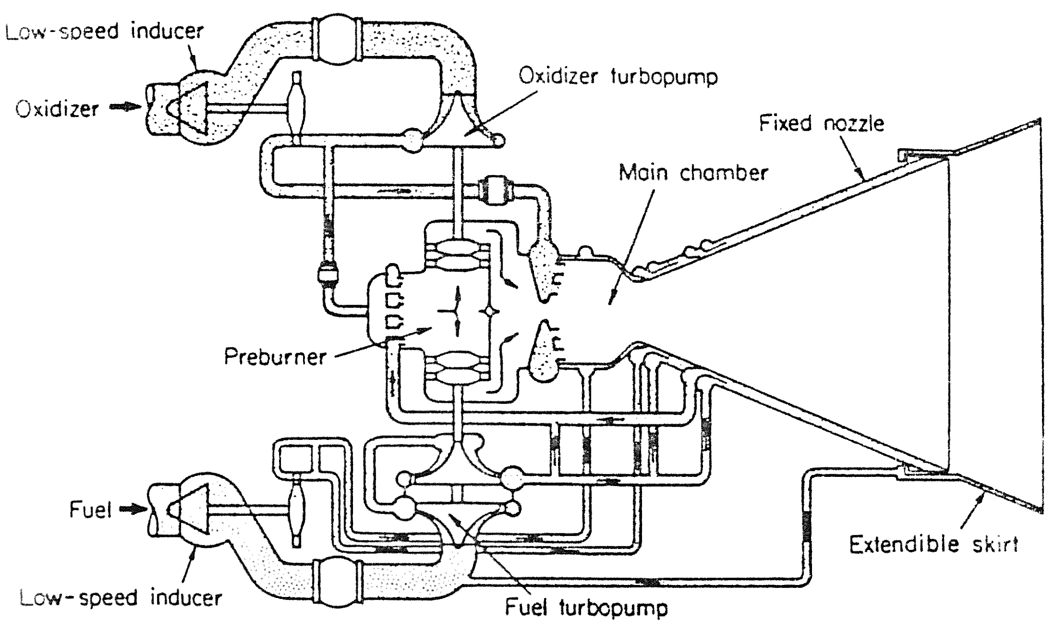


Figure 7.1-13b.- Example of multistage combustion rocket engine with variable nozzle expansion ratio (extendable skirt).

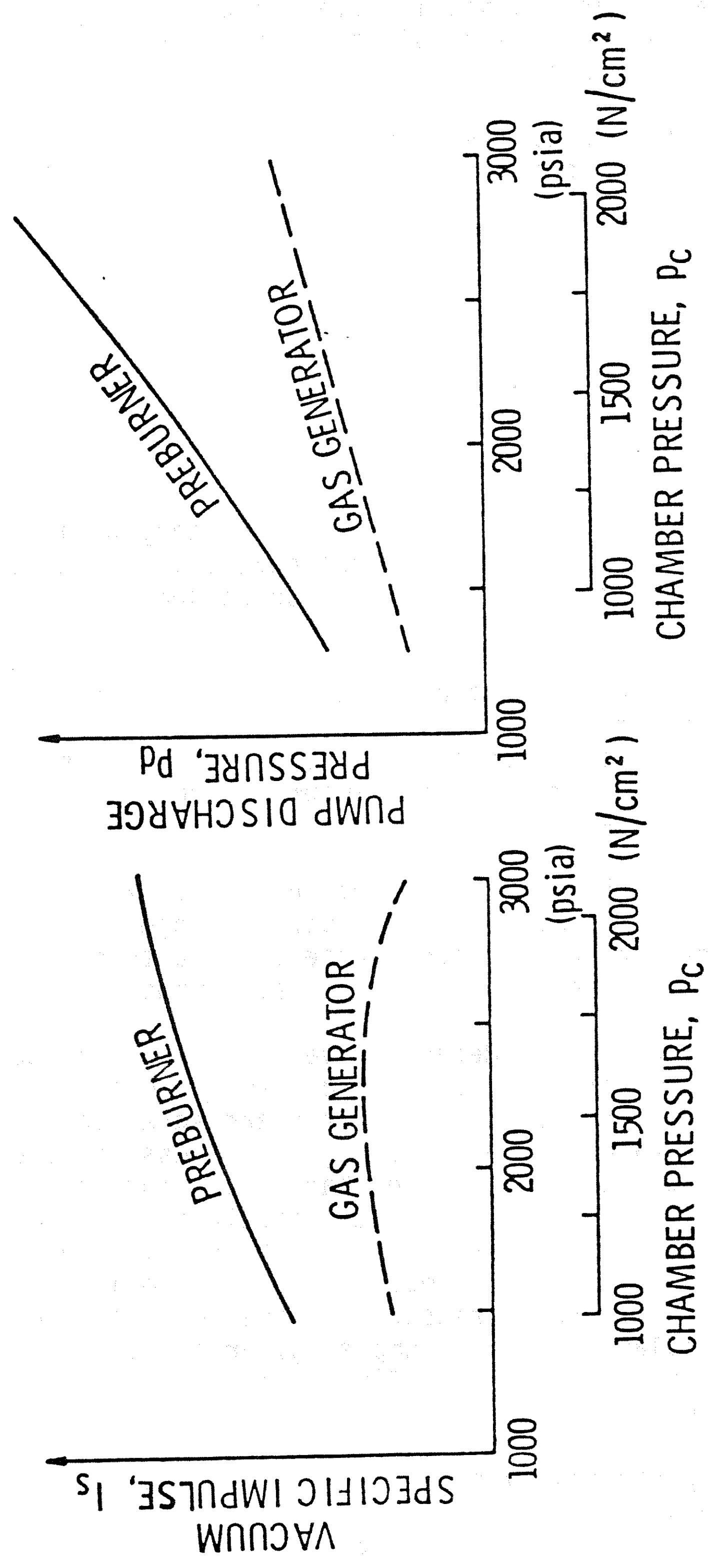


Figure 7.1-14.- Single-stage combustion with gas generator versus multistage combustion with preburner.

The thermal environment in rocket combustion chambers and nozzles is extremely severe, and elaborate techniques need to be used to maintain structural temperatures at an acceptable level (fig. 7.1-15). Heat transfer to the case wall is passed on by a combination of convective, conductive, and radiative heat transfer processes.

There are seven basic techniques that are used, either uniquely or in combination with other techniques, to maintain adequate wall temperature. These are:

- Refractory wall
- Regenerative cooling
- Transpiration cooling
- Ablation cooling
- Film cooling
- Heat sink
- Radiation cooling

7.1.5.4.1 Refractory wall.- In the case of a refractory wall combustion chamber and nozzle, structural materials that have a large resistance to high temperatures are used to avoid complication caused by using complex cooling techniques.

As a result of the lower operating temperatures, propellant exhaust velocities (specific impulses) typically are much lower than available with methods that allow cooling of combustion walls, and hence greater combustion chamber energy release. As a result, the amount of rocket propellant required for a mission is very high.

Lower temperatures can be obtained by mixing fuels and oxidizers at a mixture ratio far from stoichemetric or through the injection of relatively inert liquid or gas into the combustion chamber. An alternate method is to use propellants that give a low energy release at combustion.

7.1.5.4.2 Regenerative cooling.- Regenerative cooling techniques work by providing channels of fuel or oxidizer around the hot metal structure to draw away heat. Typically fuels are far better and safer coolants than oxidizers. The coolant flow, pumped by ullage gas pressure head or turbopumps into a jacket around the combustion chamber and nozzle, is heated and then sent into the primary combustion process itself (fig. 7.1-16). Most coolants can sustain only a relatively small heat input, otherwise they would be unsuitable as coolants as a result of transitioning into the gaseous phase, generating extreme pressures in the coolant lines, leaving film deposits in the coolant lines, decomposing, or exploding. In most cases, it is necessary to keep the bulk fluid temperature below the boiling point although localized boiling may be possible.

One disadvantage of regeneratively cooled systems is that the pressure drop experienced to provide adequate cooling results in higher pressure requirements for turbopump or gas pressurized propellant feed systems.

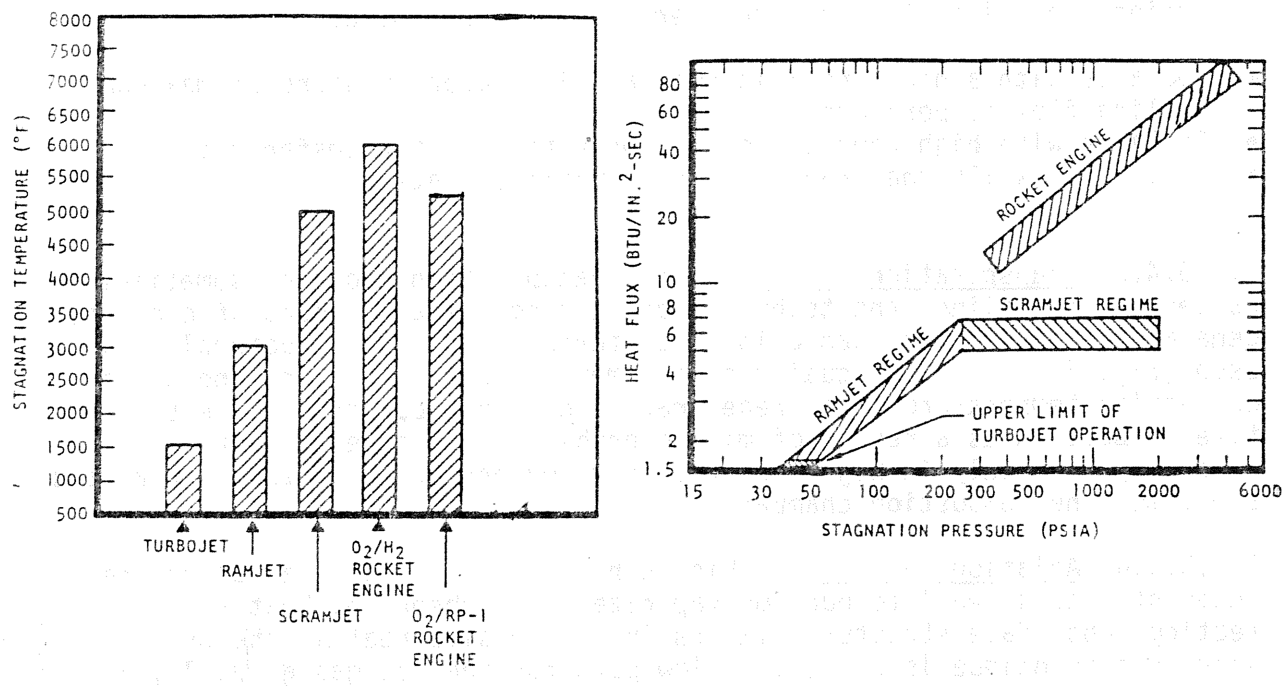


Figure 7.1-15.- Rocket engine heat transfer.

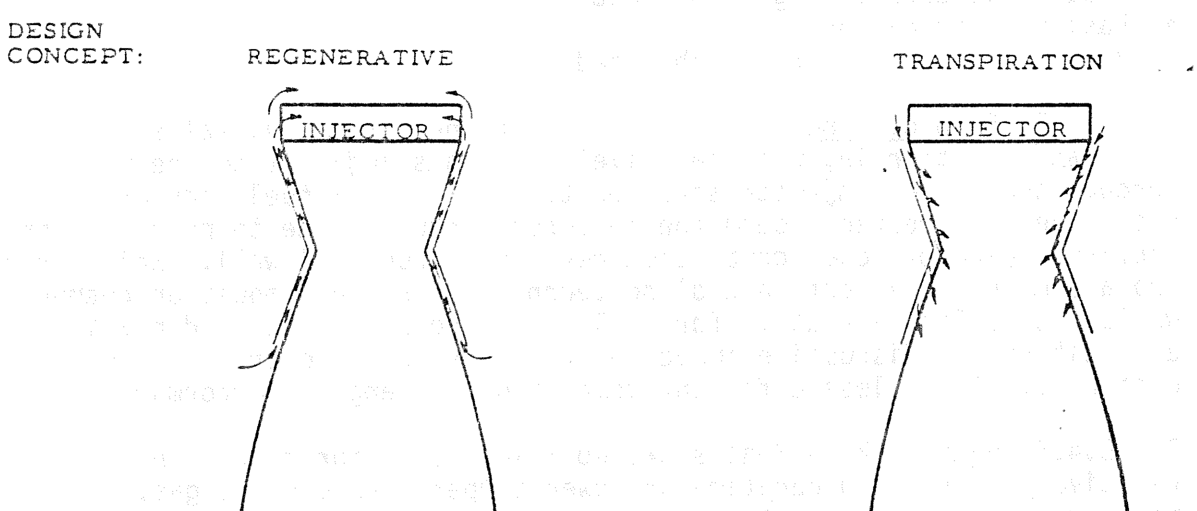


Figure 7.1-16. - Regenerative and transpiration cooling.

Qualities that allow for good regenerative cooling include:

- Coolants with a high specific heat and low vapor pressure at maximum cooling flow temperature
- Coolants with high conductive and convective heat transfer rates
- Structural walls that have high heat transfer rates

7.1.5.4.3 Transpiration cooling. - In transpiration cooling, sometimes called sweat cooling, the technique of sending a small amount of coolant, generally fuels rather than oxidizers, through a porous structural wall is used (fig. 7.1-16). It usually allows more effective cooling and higher combustion temperatures than regenerative cooling, but creates less effective combustion as a result of mixing problems. As a result, it is generally used only in high heat transfer regions of the nozzle throat and sometimes the combustion chamber.

7.1.5.4.4 Ablation cooling. - This technique uses a material along the inner structural wall to burn or vaporize away, absorbing heat and protecting subsurface structural layers from excessive heat in the process. When this technique is used, the flow path for the hot gas gradually expands in diameter over the duration of the engine burn, slightly changing thrust and performance conditions. Carbon composites have been widely used as ablatives in this role. Qualities for good ablatives are:

- Low thermal conductivity
- High heat absorption before vaporization
- Good resistance to high temperatures
- Ease of fabrication
- Resistance to wear from high speed flow

7.1.5.4.5 Film cooling. - Film cooling of the structural walls is obtained by creating a thin layer of relatively cool gas adjacent to the walls through the proper injector spray pattern. Either a fuel rich or oxidizer rich layer is created around the injector circumference to produce a protective region of lower combustion near the structural wall. While providing a simple and effective cooling technique for the combustion chamber walls, it is less effective for cooling the nozzle throat and nozzle bell as a result of the disruptive impact of boundary layers over long flow distances. Also, less efficient combustion and engine performance exist.

7.1.5.4.6 Heat sink. - Heat sinks work well only for rockets having a relatively short burn duration or lower temperature exhaust gas. This technique provides a jacket of material in the combustion zone wall that can absorb large amounts of heat quickly without leading to structural failure. Materials that are good heat sinks have good heat conductance, high specific heats, and a wide temperature range over which structural strength exists.

7.1.5.4.7 Radiation cooling. - Another option to provide cooling is radiation heat transfer away from the rocket engine to deep space or planetary/lunar background. This technique works best with engine nozzles rather than turbopumps or combustion chambers as a result of thinner walls for improved heat conduction and larger surface area for more radiative heat transfer. Radiative heat transfer works best with materials having high thermal surface emission characteristics and high heat conductivity. As thermal radiation emission (Btu's/sec) is a function of absolute temperature to the fourth power, small temperature changes can result in large heat transfer increases. Doubling absolute temperature (degrees rankine/kelvin) can increase heat transfer by a factor of 16 when radiating to deep space with a near absolute zero background temperature.

7.1.5.5 Control Valves

Control valves are used to accomplish a wide variety of tasks in liquid rocket propulsion systems, each with their own requirements. The functions that require control valves include:

- Regulating fuel and oxidizer flow to the combustion chamber and turbopumps
- Controlling the flow of propellant tank pressurizing gas and allowing for relief valve protection
- Providing for nozzle/combustion chamber coolant flow
- Inert gas purges
- Fuel and oxidizer dumps and blowdowns
- Thrust vector control systems
- Propellant fill and drain and safing operations
- Rocket staging and separation systems

Valves can be designed to be preset-loaded closed, powered open; preset-loaded open, powered closed; or bistable, where the valve will remain in its last commanded state unless powered in the other direction.

There are seven basic types of control valves that can be used in rocket engine systems. They can operate independently or in conjunction with other systems. They are:

- Hydraulic systems
- Pneumatic systems
- Electromechanical systems
- Action envelopes
- Thermoreactive systems
- Mechanical systems
- Pyrotechnique systems

7.1.5.5.1 Hydraulic systems. - Most hydraulic power systems for control valves operate using a piston and transfer valve and have characteristics that are easily derived once flow rate, piston dimensions, friction coefficients, orifice size, and mass and acceleration loadings are known.

The advantages of hydraulic systems include low weight per unit when power is needed, low inertia of output member, high operating speed, large force development, and precise positional control.

The disadvantages include the need for a remote power source, generally an auxiliary power unit (APU); high pressure lines, typically 1000 to 8000 psia, running to control valves with possible leakage and temperature control problems over long lengths; short time operating constraints because of the need for consumables for APU's imposing weight constraints on rocket payloads; and high temperature exhaust gases from APU's, unless alternate power sources exist.

7.1.5.5.2 Pneumatic power systems. - Pneumatic systems generally use piston and control valves like hydraulic systems and are connected to high pressure tanks of inert gas, typically 2000 to 5000 psia, such as nitrogen or helium.

The advantages of pneumatic systems include very fast operation time; low numbers of transfer lines, as the gas can be vented to space or atmosphere rather than require return lines as in hydraulic systems; good load carrying capacity; and low inertia of components.

Disadvantages include possible line leakage; more easily effected by contamination problems than other systems; consumables use and storage if orificed vents are used with valves rather than return lines; limited operation time due to consumables usage; and less precise positional control than hydraulics as compressible gases are used.

7.1.5.5.3 Electrical control valves. - Electrical powered systems are used in many applications using either motors or solenoids for valve operation.

The advantages of electrical systems include easily powered without using pressurized lines; good acceleration and velocity control; good reliability; and precise positional control.

Disadvantages are power source required which may or may not involve consumables usage, and high weight per unit power provided.

7.1.5.5.4 Action envelopes. - Action envelopes are a relatively new option for valve control and can be thought of as a subset of hydraulic or pneumatic systems. The design of action envelopes involves an expandable bellows system filled by a fluid (figure 7.1-17). As fluid pressure builds, the bellows expands and contracts lengthwise. As pressure decreases, the side walls of the bellows use wall tension to become narrower and increase in length. The overall behavior resembles that of human muscles.

Advantages include sensitive control, use over a wide range of power levels required, and simple operation.

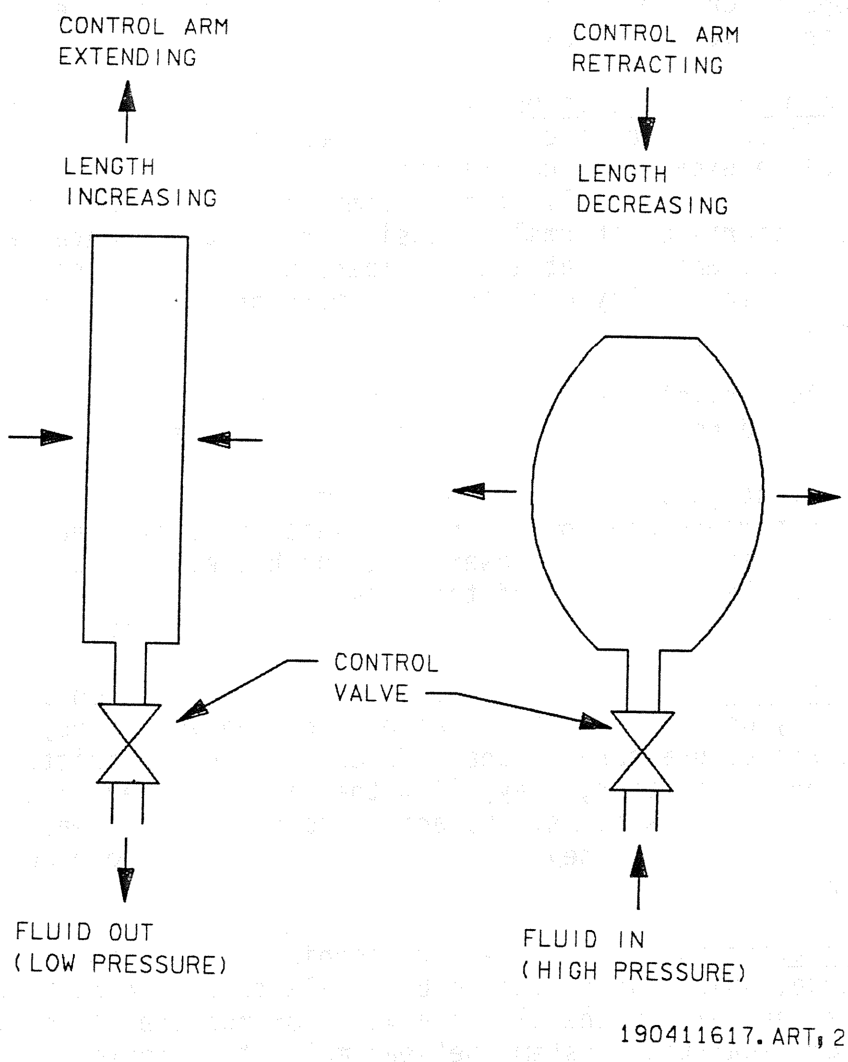


Figure 7.1-17.- Action envelope design.

The disadvantages are similar to those listed for hydraulic or pneumatic systems, depending on type of fluid used. Also, there are relatively few suppliers of this type of hardware.

7.1.5.5.5 Thermoreactive systems. - These systems use a type of structural material that is sensitive to changes in temperature to produce conformal changes that cause expansion, contraction, or bending that is coupled to a control valve. It could involve two dissimilar metals bonded together with different coefficients of thermal expansion or a type of material that undergoes a phase transition at a set temperature, such as some nickel-titanium alloys. This alloy can store a memory of a prior structural configuration in its grain structure.

Advantages include simple, reliable designs; lack of active control system; no consumables; and no power or high pressure lines needed.

The prime disadvantage of this type of system is it can offer controlling action based on temperature inputs only, unless an active heating/cooling line is provided, in which case power lines or heated/cooled fluid lines would be needed. Reaction time of the valve is relatively slow as it depends on material specific heats and thermal conductivity.

7.1.5.5.6 Mechanical systems. - These valves are controlled by springs and are most commonly used as relief valves or check valves. They are preset to one position, and as pressure on one side changes to a predicted level, depending on spring constant, they allow the valve to open or close. They are simple and reliable systems. No active control is allowed in a purely mechanical valve, although they are quite often coupled to other active control systems.

7.1.5.5.7 Pyrotechnique systems. - Pyrotechnique valves consist of a small explosive charge, which, when ignited by an electrical charge, causes a high pressure gas surge which opens/closes a valve or ruptures a bolt. They are widely used for separation systems between multistage rockets, range safety self destruct systems, releasing the vehicle from ground holddown posts, and also for single event valves, which are designed to function once during a mission, such as fuel or pneumatic isolation valves. There are some pyro valves, however, that can be used in a dual action mode to operate more than once.

Pyrotechnique valves have extremely quick reaction times, are very reliable, and can be scaled to a wide variety of valve and component sizes. The major problems associated with pyro valves and bolts are occasional fragmentation debris escaping the valve containment shells and shock/vibration loads during actuation, damaging nearby electrical systems and puncturing thin walled storage vessels (e.g., propellant tanks) in the vicinity.

7.1.5.6 Nozzles

7.1.5.6.1 Nozzle theory and operation. - The purpose of the rocket nozzle is to convert the energy released by the chemical reactions in the combustion chamber into kinetic energy at the nozzle exit plane as efficiently as possible. As explained in section 7.1.3, the thrust of a rocket is maximum when the nozzle exit plane pressure equals the ambient pressure. At this condition, the thrust equals the product of the mass flow rate multiplied by the exit velocity of the exhaust gases. To maximize exhaust velocity, nozzles first accelerate flow from the combustion chamber to sonic velocity through the use of a convergent nozzle. The Mach number is ideally equal to 1.0 at the nozzle throat. After passing through the throat, the flow accelerates to supersonic speed in the diverging part of the nozzle. As the flow accelerates in the nozzle the static gas temperatures and pressures drop, as shown in figure 7.1-18. This occurs as a result of thermal energy of molecular motion getting converted into kinetic energy.

If the static pressure is allowed to drop to ambient pressure at the nozzle exit plane it is perfectly expanded and offers the highest thrust per unit of mass flow available. More commonly, however, the flow will be either overexpanded or underexpanded (fig. 7.1-19). Both of these cases will result in less nozzle efficiency and greater propellant consumption.

In overexpanded flow, the exit plane pressure that the nozzle should reach is less than the ambient pressure, and a shock wave front occurs inside the nozzle bell. The shock wave angle is determined by the Mach number of the exhaust flow and exit plane pressure ratio ($P_{ambient}/P_{exit}$).

In underexpanded flow, the exit plane pressure is greater than the ambient pressure. In this case, expansion waves reflect from the nozzle exit cone. These expansion waves reflect off the free jet boundary as compression waves and may converge to form a shock.

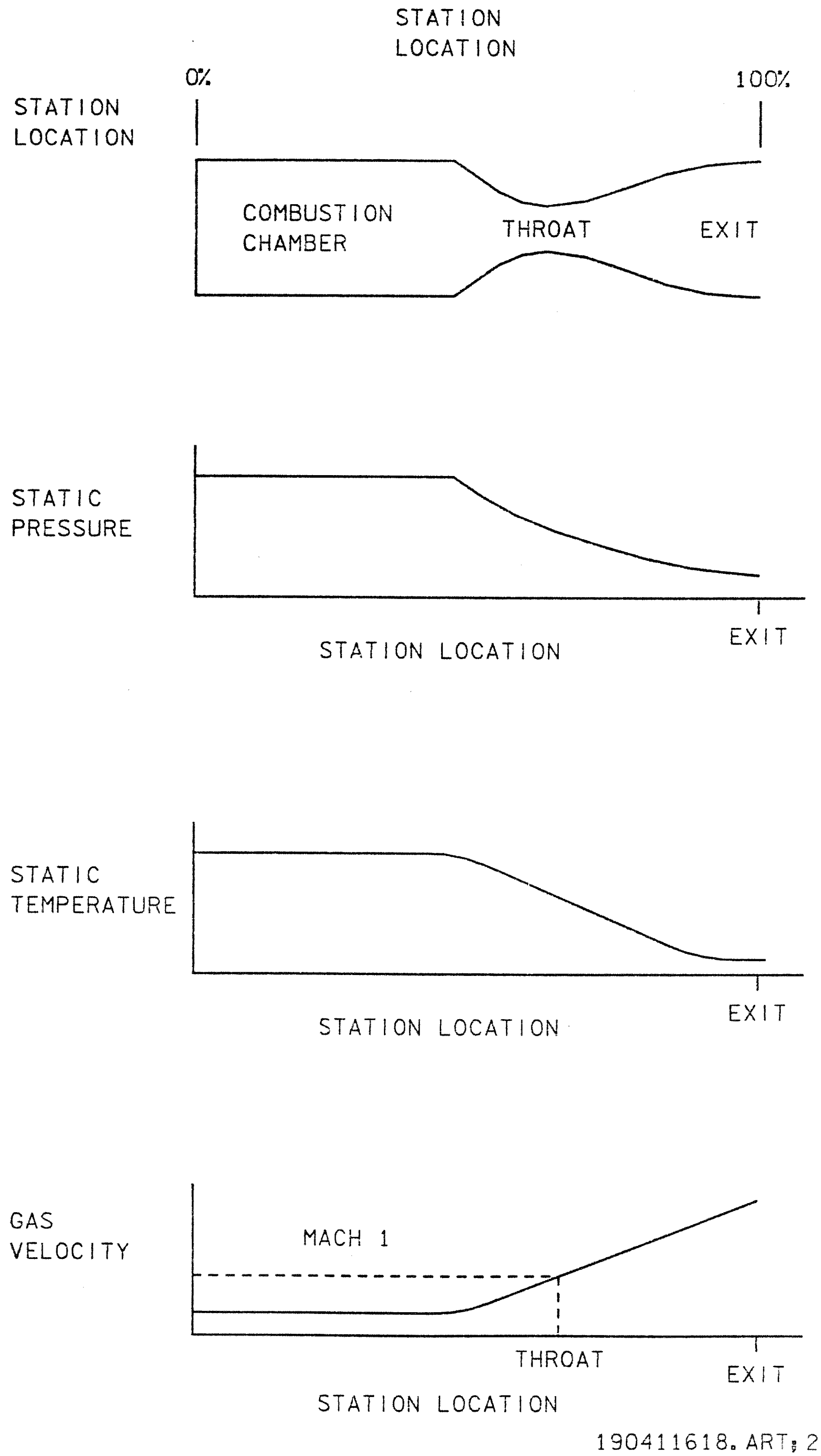
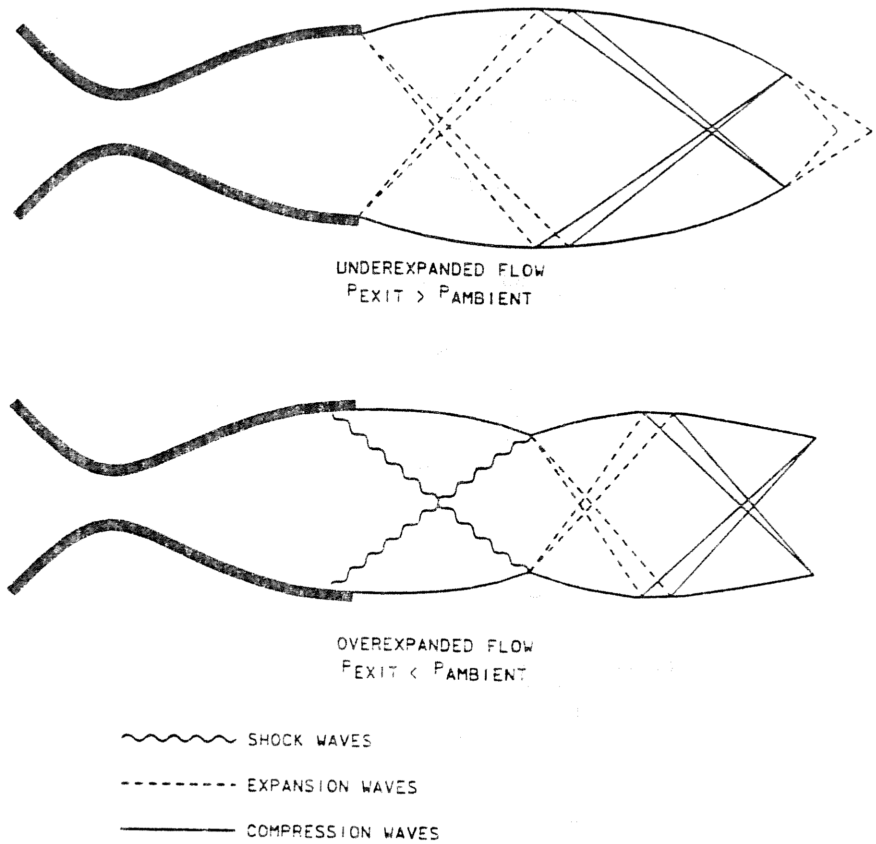


Figure 7.1-18.- Typical nozzle flow characteristics.



(refs. 4 and 17)

Figure 7.1-19.- Nozzle performance characteristics.

For fixed geometry bell or cone nozzles that are designed to operate through a range of ambient pressures, such as first stage engines in an Earth to orbit vehicle, nozzles are quite often designed to operate at an intermediate pressure ratio. That is, they will have overexpanded flow at low altitudes and underexpanded flow at high altitudes.

7.1.5.6.2 Nozzle design options. - There are basically four types of nozzle design configurations. These include:

- Standard fixed geometry bell or cone
- Plug nozzle or aerospike
- Expansion-deflection nozzle
- Variable geometry

Figure 7.1-20 illustrates the shape of these four configurations. Figure 7.1-21 shows specific impulse as a function of altitude for two nozzle expansion ratios. The advantages of these four configuration are shown in table 7.1-VI.

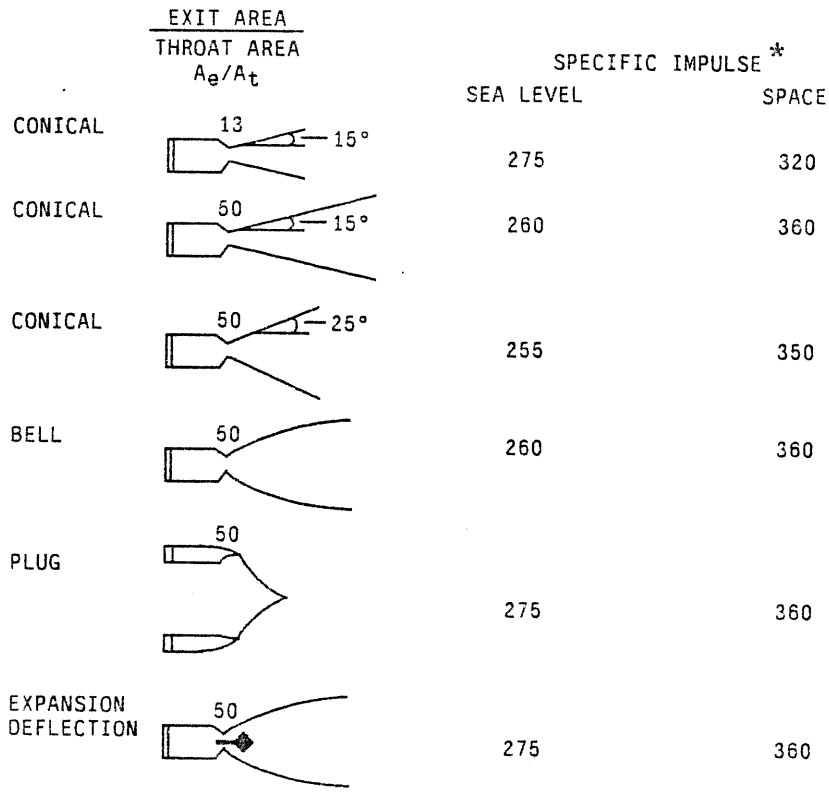


Figure 7.1-20.- Nozzle configurations.

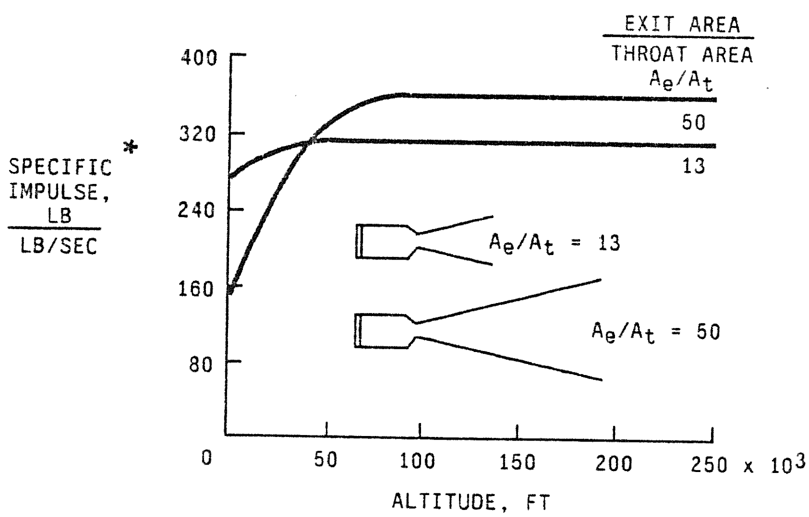


Figure 7.1-21.- Nozzle performance versus altitude.

*Assumes constant power setting with overexpansion at sea level, under-expansion in space.

TABLE 7.1-VI.- NOZZLE DESIGN CONFIGURATIONS

Design	Advantages
Standard fixed geometry bell or cone	• Well proven design and used in virtually all rocket motors ever built • Simple construction • Cone is simpler than bell, but not quite as efficient • Easier to support and cool than plug nozzle or variable geometry nozzle • Not as wide per unit thrust as expansion deflection nozzle • Works well in a vacuum or with high combustion chamber pressures • Much less complex and cheaper than variable geometry nozzle
Plug nozzle or aerospike (fig. 7.1-22)	• Free stream jet boundary allows exit plane pressure to equal ambient pressure for perfect expansion. No over or under expansion and higher thrust per unit mass of propellant. • Shorter length per unit thrust than standard bell • Flow pattern can be controlled around nozzle rim periphery to produce differential thrust rather than using nozzle gimbaling for thrust vectoring • Works particularity well with first stage ascent rockets with low chamber pressures or high ambient pressures
Expansion deflection nozzle	• Free stream jet boundary will allow avoidance of under or over expansion problems similarly to plug nozzle. Internal free stream jet boundary, rather than external free stream boundary as in plug nozzle • Shorter length per unit thrust than standard bell • Works particularly well with first stage ascent engines with low chamber pressure or high ambient pressures
Variable geometry nozzle (fig. 7.1-13b)	• Allows for variable expansion ratio and higher specific impulse for variable back pressure • Shorter length of nozzle for given thrust than fixed geometry - nozzle extensions can telescope or fold over on top of each other

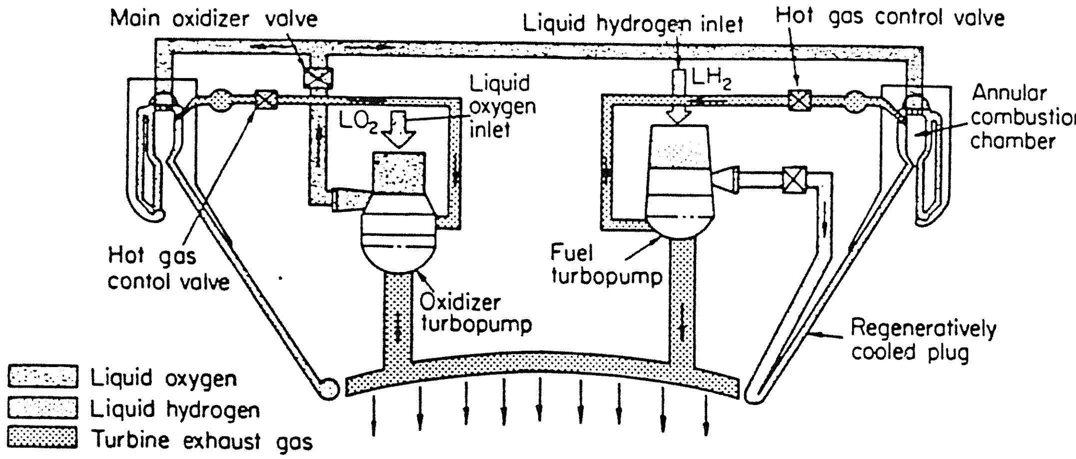


Figure 7.1-22.- Aerospike nozzle.

7.1.5.7 Thrust Vector Control

Guidance of rockets requires some type of system to control roll, pitch, and yaw rates of the vehicle over the flight path. The options include (fig. 7.1-23):

- Control fins in ambient air
- Nozzle gimbaling
- Secondary flow injection
- Exhaust gas control vanes
- Differential engine thrust

7.1.5.7.1 Control fins in ambient air. - Without thrust vector control, rockets can use aerodynamic forces on movable fins quite effectively in high speed flight at lower altitudes. Once higher altitudes are reached, other mechanisms must be used.

7.1.5.7.2 Nozzle gimbaling. - Most rockets use some type of gimbaling for thrust vector control. This usually involves dual hydraulic actuators applied to a nozzle on a high strength gimbal bearing. Flexible, bellows like ducting is used to provide propellant flow across the movable interface.

7.1.5.7.3 Secondary flow injection. - High pressure injection of gases or liquids into the nozzle can create shock waves which can vector the flow of the exhaust gases in a way to produce vectored thrust. This eliminates the need for hydraulic actuators and flexible bellows, but causes a drop in specific impulse while vectoring is active.

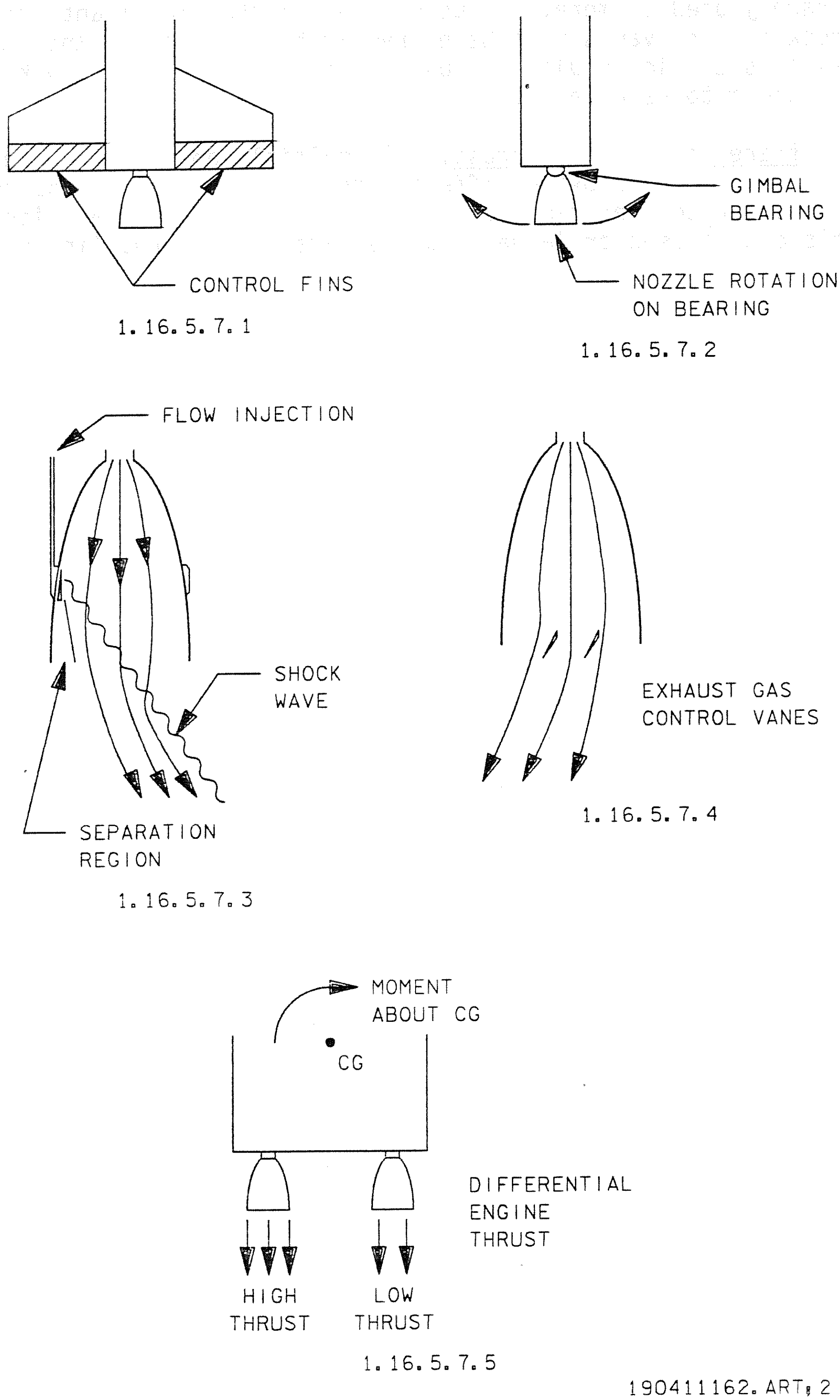


Figure 7.1-23.- Thrust vector control.

7.1.5.7.4 Exhaust gas control vanes. - This mechanism, used in the V-2 rocket, is rarely used anymore. It uses vanes in the propellant exhaust to steer the rocket. The vanes must be extremely heat resistant and impose minimal loss in specific impulse through frictional effects and wakes in the nozzle flow stream to be effective.

7.1.5.7.5 Differential engine thrust. - If multiple engines are used, this mechanism can be used to provide different thrust levels about the rocket centerline to steer the vehicle. Either multiple small engines with on/off thrust levels can be used or large thrust engines with throttling capability can be used.

7.1.6 References

1. Encyclopedia Americana, International Edition, Grolier Incorporated, Danbury, Conn, 1989.

2. Wernher von Braun & Frederik Ordway: History of Rocketry & Space Travel. Thomas Crowell Co, New York, NY 1966.

3. Michael Stoiko: Soviet Rocketry. Holt, Rinehart, and Winston, New York, NY 1970.

4. The New Encyclopedia Britannica, 15th edition, Encyclopedia Britannica Inc, Chicago, Illinois, 1985.

5. Brian O'Leary: Space Industrialization. CRC Press, (Boca Raton, Florida), 1982.

6. David Harrje And Frederick Reardon: Liquid Propellant Rocket Combustion Instability. NASA SP-194, (Washington, D.C.), 1972.

7. Space Systems Technology, SP-593. Society Of Automotive Engineers, Inc., (Warrendale, PA), 1984.

8. NASA Historical Data Book. Programs And Projects, 1969-1978.

9. Julian LeVine & Neville Barter: TRW Space Log. TRW Space and Technology Group, (Redondo Beach, CA), 1990.

10. This is NASA. NASA EP-155, Johnson Space Center, (Houston, TX), 1979.

11. Shuttle Operational Data Book. Vol. I, JSC-08934, Rev E, Johnson Space Center, (Houston, TX), 1989.

12. Handbook Of Satellites and Space Vehicles. D. Van Nostrand Company, Inc., (Princeton, N.J.), 1965.

13. Heinz Hermann Koelle: Handbook of Astronautical Engineering. McGraw-Hill, Inc., (New York, N.Y.), 1961.

14. Boris Kit & Douglas Evered: Rocket Propellant Handbook. MacMillan Company, (New York, N.Y.), 1960.

15. Stanley Sarner: Propellant Chemistry. Reinhold Publishing, (New York, N.Y.), 1966.

16. Dieter Huzel & David Huang: Design of Liquid Rocket Propellant Engines. NASA SP-125, NASA Headquarters, (Washington, D.C.) 1971.

17. Phillip Hill And Carl Peterson: Mechanics And Thermodynamics Of Propulsion. Addison-Wesley Publishing, (Reading, MA) 1970..

18. Proceedings of The NASA-University Conference On The Science And Technology of Space Exploration. Vol. 2, NASA SP-11, (Cleveland, OH), 1970.

19. Space Shuttle Main Engine, RI/RD87-142, Rockwell International, Rocketdyne Division, (Canoga Park, CA), 1987.

20. Space Transportation System Technology Symposium. Vol. 4, NASA TM-X-52876, NASA Lewis Research Center, (Cleveland, OH), 1970.

21. Edward Safford, Jr.: Handbook Of Advanced Robotics. Tab Books Inc., (Blue Ridge Summit, PA), 1982.

7.2 SOLID ROCKET MOTOR FUNDAMENTALS

7.2.1 General

SRM's provide low cost, reliable access to space. SRM's are relatively economical because they avoid launch site propellant handling, minimize launch site support, provide long term standby capability and require virtually no effort for extended holds. Overall, solid boosters are less expensive than liquid boosters. Used in the Atlas, Delta, Titan, and shuttle space programs, solid boosters have demonstrated high reliability. Considered as a group, they reflect a collective reliability of 0.995 at the 95 confidence level. This reliability is as good as, or better than, that of liquid rocket motors. Solid rocket propulsion is effective for large launch vehicles because very large mass flows and hence high thrust are easily obtained with SRM's. To maximize payload success, solid and liquid propulsion are sometimes mixed. Combining the two minimizes system development and operational costs and, by utilizing the advantages of each, provides mission flexibility.

SRM's are chemical rockets; the energy required to accelerate the propellant comes from the propellant itself. Released in the chemical reaction is the energy to be used for thrust. SRM's are simple in structure, consisting of three major parts: a propellant supply and feed system, a combustion chamber, and an exhaust nozzle.

7.2.2 Propellant

SRM's contain a solid block of propellant, which includes all of the material necessary for the chemical reaction. The block of propellant, or grain, can be of two types, homogeneous or composite. In homogenous propellants, all of the materials are contained within the same molecule. Double-based propellants and combinations of nitroglycerin-nitrocellulose are examples of homogeneous propellants. Small quantities of additives improve the burn of the grain. Potassium salt promotes smooth burning at low temperatures. Diethypthalate is an auxiliary plasticizer that improves mechanical properties. Ethyl centralite counteracts autocatalytic decomposition of major constituents. This function is especially important, because if the decomposition goes unchecked, it could lead to the self-ignition of the propellants. The addition of carbon black prevents the transmission of radiant energy that might cause internal ignition around internal voids or impurities. Candelilla wax is added to propellant formed by an extrusion process. To improve performance, metallic powders such as aluminum and additional composite oxidizers are added.

Composite propellants are mixtures of oxidizing crystals in an organic, plastic-like binder. The oxidizers are usually ground crystals of potassium, lithium, or ammonium perchlorate or nitrate. Because in most of these oxidizers there is low available oxygen and the combustion temperatures are relatively low, composite propellants have high oxidizer contents. Because oxidizers are more dense than fuels, a high oxidizer content increases the density of the propellant. Denser propellant is good because it requires a smaller combustion chamber. However, oxidizers tend to have poorer castability and mechanical properties. Generally these drawbacks can be offset by choosing suitable additives and usually partially oxidized fuels. The mechanical characteristics of the propellant are directly related to the quantity of plastic binder and fuel. Possible fuels or binders are asphalt, rubber, synthetic resins, and elastomers.

7.2.3 Combustion Chamber

The combustion chamber is a high-pressure tank designed to generate high-temperature and high-pressure combustion products. The chamber contains the solid propellant and sufficient void space to permit stable combustion. The combustion proceeds from the grain surface. Affecting the design of the combustion chamber are the engine performance parameters burn rate, burn rate exponent, area, and pressure.

7.2.3 Burn Rate

Burn rate is a function of the velocity of the gaseous products over the surface of the solid, and the temperature and pressure in the combustion chamber. The surface combustion process requires heat transfer from the combustion products to the solid material to vaporize the solids. High fluid velocity over the grain surface can substantially increase the heat transfer rate. Velocity varies appreciably within the void space. As a result the burn rate can vary significantly along the grain length, thus complicating the grain design problem. The effect of increased burnable rate with increasing combustion-product velocity is called erosive burning, although the effects of heat transfer are much more important than any erosion of the solid material.

The burn rate is proportional to the combustion chamber pressure raised to the burn rate exponent. The constant of proportionality is a function of the initial temperature of the propellant. Variation of burn rate with temperature can be dangerous; too high a burn rate an cause excessive vehicle acceleration or chamber failure, and too low a rate produces insufficient thrust. Use of composite propellants reduces temperature sensitivity.

7.2.3.2 Burn Rate Exponent

The burn rate exponent must be less than one for stable combustion. This is because the nozzle flow rate, which is directly proportional to the combustion pressure, is equal to the gas generation rate, which is proportional to the burn rate and hence the combustion pressure raised to the burn rate exponent.

7.2.3.3 Area

To maintain constant combustion pressure and thus constant thrust, the burning area must remain constant. Any variation of the burning area with time depends on the burn rate and the initial geometry of the grain. A larger burning area delivers more thrust. Increasing or decreasing the burning area causes progressive or regressive burning and hence a rise or fall in thrust. High-thrust rockets are usually of the internal-burning type to maximize burning area. Low-thrust rockets have solid, cylindrical grains that burn only on the end surface.

7.2.3.4 Pressure Limits

Combustion is unsteady or ceases completely below a combustion limiting pressure. Therefore, it is important that grain burning is completed instantaneously along the length, because nonuniform burn-through may reduce the chamber pressure below this limit. Extinguishing prematurely may cause waste of unused propellant or overheating of exposed portions of the chamber wall. Axial variations of flow area and thickness and shape of the grain can compensate for nonuniform burning. Excessive pressures may violate the pressure limit, above which combustion is erratic and unpredictable.

7.2.3.5 Igniter

When nonhypergolic propellants are used, a pyrotechnic igniter is needed. A small chemical charge burns vigorously for a few seconds after being ignited by an electric filament. It is a one-shot ignition; there is no restart. Solid rockets require additional equipment to start, stop, and control ignition. To ensure uniform recession of the burning face, a large surface area is ignited. At the propellant surface, ignition is the result of a combination of convective and radiative heat transfer and impingement of hot particles from the igniter. Therefore, the igniter must provide sufficient surface heat and a chamber pressure above the combustion limit. (If the pressure is too high, the solid propellant may crack, leading to chamber rupture.) This initial pressure is usually provided by the gas generation of the igniter, but for high altitude starts, the nozzle is obstructed with a plug that blows out upon ignition. To terminate thrust, suitable ports or the nozzles themselves are blown off, the chamber pressure falls, and the flame is extinguished.

7.2.3.6 Stress-strain

Although the combustion chamber is merely a pressure vessel, the design is more complex. Ports for the nozzle, igniter, blowout port, and external structural members cause stress concentrations. The stress loading on the grain is due to axial pressure distribution, viscous shear from gas flow, and vehicle acceleration. The result of these stresses tends to move the propellant toward the nozzle. Opposing this stress is the support system or the wall shear force. As a result, the propellant bulges near the nozzle, reducing the port area and therefore accentuating the pressure and viscous stresses. If Young's modulus for the propellant is below a critical value, the grain will continue bulging and the pressure will continue rising until the chamber bursts. The mechanical properties of the propellant are heavily dependent on temperature. The stress-strain curve of the propellant is a function of temperature and strain rate. During curing and storage, there is a low strain rate. At ignition, the strain rate is high. High temperatures cause "slump" in the propellant, especially under long periods of storage. Low temperatures cause the grain to become brittle and cracking may occur, thus exposing a much larger burning surface than intended. In most cases, the grain is bonded directly to the case or a thin layer of inhibitor. The propellant itself is an insulator; therefore the chamber walls are kept cool. The direct bonding eliminates internal structures and the grain is adequately supported up to burnout. (For very large grains, supports may be required.) The walls are thinner and a higher fraction of the chamber volume is filled with propellant, thus the combustion chambers can be lighter.

Because the combustion temperatures are higher than the melting points of many metals and alloys, highly stressed materials must be used in the booster. To alleviate some of the thermal stresses, the chamber is cooled. One way to cool the chamber is to surround the nozzle walls with a mass of metal to absorb the heat from the hot surface. The heat capability, allowable temperature of the material, and the heat transfer rate limit the time during which this fix is effective. The chamber can also be cooled by the vaporization or sublimation of material from the inner surface of the chamber wall. The injection of liquid or gases through porous walls is called "sweat cooling." Intentional loss of wall material is called "ablation cooling." "Film cooling" uses the layer or relatively cool gas near the walls.

7.2.4 Combustion Instabilities

Particularly in large boosters, pressure oscillation within the chamber can cause combustion instabilities. Such instabilities result in pressure buildup and increased heat transfer, which can cause the engine to fail. The specific cause of these instabilities is not fully understood because the problem occurs only on the full-scale engines, not on the models. The energy released in combustion is rather uniform. Because the combustion rate influenced by the propellant pressure and velocity varies, the propellant pressure and velocity patterns may interact with the energy release to produce sustained oscillations. Such oscillations produce high erosive burn rates. If the combustion geometry does not change to a stable configuration, it may lead to engine failure. Instabilities can be reduced by placing an irregular rod of non-burning material within the burning volume or drilling radial holes at intervals along the grain because the disturbance tends to bread up the wave patterns.

7.2.5 Nozzles

The main concerns in determining nozzle shape are friction, heat transfer, changes in composition, and shocks. For the converging portion of the nozzle, any reasonably smooth contour gives good subsonic flow. In the diverging portion of the nozzle, improper shaping can cause shock resulting in performance loss. The nozzles are generally simple, conical diverging sections with half-angles in the range of 12° to 18°. Angles that are too large cause radial flow losses and flow separation from the walls. Contouring the nozzle to give a uniform axial exhaust velocity eliminates radial flow losses. Angles that are too small require excessive nozzle length and therefore cause large friction losses.

7.2.5.1 Boundary Layer Effects

The propellant flow is altered by viscous effects near the boundaries. Fortunately, the pressure gradient is favorable and the boundary layers are thin. The boundary layer affects the performance in three ways. First, the boundary layer alters the free-stream characteristics of the flow. At the throat, the boundary layer displacement thickness reduces the flow area and hence the mass flow rate. Second, the boundary layer affects the rocket thrust directly through the shear stress or skin friction on the nozzle wall, since thrust is simply the integrated stress over the thrust chamber surface. The thrust is the result of the pressure stress vectors over the thrust chamber surface. For typical rocket propellants, viscous stresses are very small relative to pressure stresses so that viscous contributions to thrust are usually negligible. Third, under ambient pressure, shocks can occur even in well-designed nozzles. The position and strength of the shock are functions of the boundary layer behavior and the free-stream and ambient conditions.

7.2.5.2 Back Pressure

Launch vehicles in particular are subject to wide variations of "back" or ambient pressure. This fluctuation can seriously alter performance. If the back pressure is less than the exhaust pressure, the flow within the nozzle will remain undisturbed. A pressure drop downstream of a supersonic nozzle cannot propagate upstream to the nozzle, because an expansion wave travels at only the local speed of sound. Therefore the pressure change occurs as a series of expansion waves beyond the exit plane. This is known as an "under expanded nozzle" because the flow has not expanded to ambient pressure.

If the back pressure is greater than the exhaust pressure, the pressure difference can disturb the flow within the nozzle. A pressure increase occurs as a sudden increase or shock propagating relative to the fluid at supersonic speed. As the pressure rises, the shock gets stronger and propagates at higher speeds. When the pressure gets high enough, the boundary layer separates upstream of the shock. The shock thus moves upstream until it reaches an equilibrium point where the pressure increase is just small enough to avoid separation upstream of the shock. The back pressure only has detrimental effects when it is sufficiently larger than the design pressure. Not all engines should be designed for large pressures because at high latitudes (low pressures) an engine designed for high pressure will not give as much thrust as one designed for the lower pressure.

7.2.6 Thrust Vector Control

The thrust chamber cannot be pivoted to achieve thrust vector control because it composes the entire body of the engine. The nozzle must be gimballed relative to the combustion chamber. This gimballing requires a joint to contain high-temperature, high-pressure propellants and at the same time be reasonably easy to move. In old rockets, movable vanes were immersed in the exhaust flow. Non-axisymmetric injection of secondary fluid in the nozzle has also been tried, but it tends to disrupt the supersonic nozzle, causing shocks with the nonaxial exhaust momentum flux.

7.2.7 References

Hill, Philip G. and Peterson, Carl R.. Mechanics and Thermodynamics of Propulsion. Addison-Wesley Publishing Company, Reading, MA, 1965.

"Low Cost Reliable Access to Space: A Solids Approach" (from working group from solid rocket industry--Elliott Mitchell).

7.3 STORAGE OF PRESSURIZED GASES AND CRYOGENIC FLUIDS

7.3.1 Introduction

The space shuttle employs a variety of gases during its operation. The uses of these gases range from purges for the propulsion systems to the fuel burned in the main engines. There are many hazards associated with the use of these chemical elements, such as storing and handling these gases. The environments encountered are extreme, dictating the design and operation of the facilities involved to ensure safety and reliability. The two methods for storing gases are at ambient temperature and high pressure, and cryogenically at ambient pressure and low temperature.

7.3.2 High Pressure Storage

Storing gases at high pressure and ambient temperatures is the easier of the two methods, but there is usually a higher weight penalty. The higher weight is a result of the strength requirements the storage container must satisfy. Ideally, this type of pressure vessel would be ductile enough that in the event of overpressurization, the structure would deform and fissure rather than fragment and explode, as would be the case for a brittle structure. This system is the most reliable when high pressure is a major system requirement.

7.3.3 Cryogenic Storage

Cryogenics deals with the properties and the use of materials at extremely low temperatures. Cryogenics also includes the production, storage, and use of cryogenic fluids.

A gas is considered cryogenic if it can be changed to its liquid state by removing heat and reducing the temperature to a very low value. Although this temperature range is not precisely defined, most gases are considered cryogenic if they can be liquefied at or below -240° F. The most common cryogenic fluids include air, argon, helium, hydrogen, methane, neon, nitrogen, and oxygen.

Cryogenic storage is more applicable when large quantities of gas are required, but container space and mass are constrained. Strength requirements for such a container are substantially less since the pressures encountered are a fraction of what they would be if the gas was stored at ambient temperature; therefore, the container will weigh far less.

In cryogenic containers, the pressure is approximately at ambient. The gas is usually stored as a liquid at or slightly below its boiling point at ambient pressure. A significant advantage of the liquid state of a gas is its greater density compared to the vapor state, allowing far more mass to be stored in a given volume.

Another benefit associated with storing gas in the liquid state is the stability of the liquid as a result of controlled boil-off. This is the process in which heat is released because of the vaporization of the liquid gas, which in turn keeps the temperature of the liquid at the boiling point with great accuracy. Even in the presence of excess heat, boil-off will continue to maintain the liquid at the desired temperature.

The extremely low temperatures experienced in cryogenic storage do cause some difficulties and are an important factor in the design of the containers and their systems. All tanks and lines have to be very well insulated. This is important for the control of boil-off rates. The materials used for the tanks and lines must be very ductile at low temperatures. They also must be high in strength in the event that excessive vaporization occurs and the tank and lines become overpressurized.

7.3.4 Storage Methods in the Main Propulsion System

The space shuttle uses high-pressure, ambient temperature tanks to contain the required helium supply. The high-pressure gas is available on demand, with no need for any type of pump or pumping action. The tanks themselves are composed of a titanium liner with a Kevlar-epoxy shell. This type of tank will not fragment in the event of a structural failure.

The tanks operate at 4500 lb/in^2 and are proof-rated at 6300 psi. The helium system manifolds, however, operate at much lower pressures and therefore rely on regulators to deliver helium at lower pressures. The system also uses vent valves that relieve excess pressure in the event of a regulator or associated failure.

The space shuttle ET consists of two cryogenic containers for the liquid hydrogen and liquid oxygen. In addition to withstanding the extreme temperature environment of the propellants, the ET receives and distributes stress loads from and to the orbiter and SRB's. To meet these requirements, the tanks are constructed from several very ductile aluminum alloys and are thoroughly insulated.

The insulation is an essential element in the cryotank system. The thermal protection system helps control the liquid's temperature, which maintains the quality of the propellant as well as proper boil-off rates. The ET also has several backup systems to help control temperature and deal with excess pressure, which is the result of converting liquid propellant to gas.

The LOX tank employs a helium main feedline injection system to prevent geysers from forming due to temperature gradients. The helium bubbling through the LOX vaporizes a portion of the propellant, which in turn cools the remainder. The LH_2 tank uses a recirculation system which continuously mixes the propellant to maintain a constant temperature.

Both the LOX and LH_2 tanks are each equipped with vent/relief valves in the event of tank overpressurization. The venting and relieving are done by the same valve and their operation is automatic upon pressure build up. The LOX tank relieves at 23-35 psig and the LH_2 at 35-37 psig.

7.3.5 Liquid Hydrogen

High-purity liquid hydrogen is a transparent, colorless, odorless liquid. When it can be observed, it is usually boiling vigorously because of its low boiling point. When exposed to the atmosphere, it creates a voluminous cloud of condensed water vapor. Liquid hydrogen is noncorrosive in nature and will form combustible mixtures with oxidizers. Hydrogen gas is combustible with air over a very wide range of mixtures. Liquid hydrogen is chemically stable, but because of its low boiling point, it is physically stable only when it is stored under suitable conditions.

7.3.6 Fire and Explosion Hazards

Fire hazards always exist when hydrogen gas is present. Hydrogen air mixtures containing as little as 4 percent or as much as 75 percent of hydrogen by volume can be ignited readily. Hydrogen-oxygen mixtures are flammable over a range of 4 to 94 percent hydrogen by volume. When no impurities are present, hydrogen burns in air with an invisible flame. Hydrogen-air mixtures that are unconfined generally burn rapidly but without detonation when initiated by heat, spark, or flame. However, in confined areas, or when the ignition of the hydrogen-air mixture is caused by shock source equivalent to a blasting cap or small explosive charge, the mixture can detonate.

An explosive hazard can exist if liquid hydrogen is contaminated with solid oxygen or solidified oxygen-enriched air that is formed by exposure of the liquid hydrogen to air or oxygen.

LOX does not burn, but vigorously supports combustion. Normally it not hypergolic with fuels. LOX will cause liquids fuels to cool and freeze if both liquids are brought together. Such a mixture of frozen fuel and LOX is shock sensitive and can react with the violence of a detonation. The hazard must be considered in fire control and preventive measures taken in connection with spills of liquid oxygen. Materials that react violently with oxygen include oil, grease, asphalt, kerosene, cloth, wood, paint, tar, and dirt.

Two types of combustion reactions are possible, depending on the conditions of mixing and ignition. Fuel and LOX may be mixed without a consequent fire, but detonation occurs upon ignition or mechanical shock excitation of such a mixture. If combustion is initiated before contact with fuel and oxygen or concurrently with contacts, flare or combustion results, accompanied by repeated explosions. The intensity of a combustion reaction depends on the type of fuel involved.

When mixed with LOX, all materials that burn are potential explosion hazards. These mixtures usually can be exploded by static electricity, mechanical shock, electrical spark, and other similar energy sources, particularly when the mixtures are frozen.

7.3.7 Liquid Oxygen

High-purity liquid oxygen is a light blue, transparent liquid. It boils vigorously at ambient conditions and uninsulated containers are usually frosted. Oxygen has no odor. In either gaseous or liquid form, oxygen is a strong oxidizer that vigorously supports combustion. The violence of some reactions involving LOX is due to the high reactive, oxygen-rich atmosphere surrounding the liquid. LOX is chemically stable, is not shock sensitive, and does not decompose.

7.4 COMMON ORBITER CONNECTORS

There are a variety of connecting joints (also known as connectors or fittings) on various subsystems on the orbiter. The terms joint, connectors, and fittings are used interchangeably. The focus of this section is to provide a general understanding of these basic joints. The introduction is followed by a general description of each joint and its associated advantages and disadvantages.

Introduction

Connectors, commonly referred to as fittings, are devices used in fluid systems for the purpose of joining lines or components and sealing the joint thus made. Connectors may be used for line-to-line, line-to-component, or component-to-component applications. All connectors, regardless of type, employ three functional elements: a seal, the connector-to-line joint, and the load-carrying structure. The distinction between connectors involves the manner by which these three functions are accomplished. In some separable connectors, there is a direct relationship between the functional elements and the mechanical parts of the connector, while in permanent connectors all three functions may be combined, as in a single welded joint.

Connectors can be broadly categorized as separable, permanent, and semi-permanent. Separable connectors have at least one mechanical joint that can be easily separated and reconnected a number of times. In addition to the mechanically separable joint, separable connectors may use a permanent (swaged, welded, or brazed) connection for joining the connector mating halves to the tube and/or component. The separable connector category includes threaded connectors and bolted-flange connectors. Permanent connectors use welded, brazed, soldered, or swaged connections rather then mechanical joints, and are used where reduced weight and reliability are more important than ease of separation. Two common types of permanent connectors are brazed and welded connectors. Semi-permanent connectors are a special class of permanent connectors, consisting of permanent joints (welded, brazed, or swaged) that can be used for a limited number of separation and reconnection cycles without requiring removal or replacement of lines, or special adaptors or fittings in making connections.

As a general rule, the number of connectors in any system, regardless of type, should be kept to a minimum. Any connector is a potential source of problems. In addition, they add to system weight and costs.

Numerous factors, often conflicting, enter into the selection of a connector for a given fluid system application. The following paragraphs discuss factors to be considered in selecting a connector.

Permanent or Separable Connectors

The first decision to be made in selecting a connector is whether the joint(s) under consideration should be separable or permanent. The basic distinction between permanent and separable connectors involves how the connector seal is affected. The seal of a permanent connector is affected by intermolecular cohesion at the sealing interface, whereas the separable connector seal involves mechanical mating, usually including local deformation of one of the sealing surfaces. The intermolecular bond affected in the permanent connector has a far higher probability of sealing than the mechanical mating of surfaces with the separable connector; however, the permanent connector seal cannot be separated easily, whereas the separable connector can be readily disconnected with a minimum of tooling and, in many instances, reused.

In addition to a higher probability of sealing because of fewer mechanical parts, permanent connectors also offer a weight advantage over most separable connectors, often second in importance only to reliability for flight systems. Other attributes of permanent connectors are a small envelope and resistance to shock and vibration.

On the other hand, the distinct advantage of separable over permanent connectors is the ease of making and breaking joints, often a very important consideration during system development, component replacement, repair, and system maintenance.

The heat required to make most permanent joints must be considered carefully relative to any possible detrimental effects on the system. Heat damage to polymeric seals in valves and thermal distortion are common problems in welded installations. Loss of strength in heat-affected areas must also be considered. Accessibility of the connectors to the necessary welding or brazing tools can be another limitation to the application of permanent connectors, depending upon the system configuration. Permanent connectors that join dissimilar materials such as titanium-stainless steel or aluminum-stainless steel present problems that can be overcome more readily with mechanical type joints. Susceptibility of brazed or welded joints to corrosion is another potential problem that must be considered in selecting permanent connectors.

a. Permanent connectors

The choice of permanent connectors offers several possible joining techniques, including welding, brazing, bonding, soldering, and swaging. Welding and brazing are the most common. Factors that influence the selection between types of permanent joints include permanence of the joint, temperature range, fluid compatibility, and suitability of the connector and/or tube materials and geometry for either welding or brazing. Although providing a permanent connection, brazing lends itself more readily than welding in making a joint than can be separated and rejoined if necessary. This is because a brazed connection can sometimes be reheated, separated, and later rebrazed. It is possible to provide welded joints with limited reusability but this adds additional weight to the joint. Such joints commonly are referred to as semi-permanent welded joints. Elevated temperature and joint strength are two of the major limitations with brazed joints, resulting from the relatively low melting points of brazed alloys. A practical upper temperature limit for most brazed joints is approximately 1000° F; the actual limit for specific applications depends on the materials used.

b. Separable connectors

Most separable connectors fall into one of two categories, threaded connectors and bolted-flange connectors.

A basic difference between threaded connectors and bolted-flange connectors is the number of tension (or tensile) members in the separable joint. Threaded connectors use a single nut, while flanged connectors use several bolts. The choice between threaded and flanged connectors is determined by line size and the amount of preloading required to establish a satisfactory seal over the entire range of connector loads. In smaller line sizes, threaded connectors have a weight advantage.

As the line size approaches 1 inch, applied torque limitations reduce the preloading ability of the threaded connector structural members, and only by increased size (and therefore increased weight) of these structural members can satisfactory preloading be maintained.

In general, a 2000 inch-pound torque limitation is used for a threaded connector because of wrench size and nut strength limitations. Thus, a practical limitation exists for the amount of torque applied to a single threaded connector and, consequently, the use of threaded connectors is limited by system fluid pressure and line size. Size and weight of the connector are other factors that place a line size limitation on the use of threaded connectors. For smaller line sizes (less than 1 inch), threaded connectors have a weight advantage over flanged connectors. A practical limitation on the use of flanged connectors in small line sizes is that in small flange sizes, the bolt becomes too small for reliable torquing. Threaded connectors are favored for use with smaller line sizes because of the relative ease of installation and the simplicity with which the tube may be joined to the connector. A disadvantage of threaded connectors is the danger of damage to the tube or line if the threaded flange is not restrained while the nut is torqued down.

7.4.1 Swage

7.4.1.1 Description

Swage connections are created by the cold formed joining of two pieces of metal to form a seal. Swaging is simply an operation that presses a piece of line (tubing) into a fitting or a fitting onto a line to form a joint.

In orbiter circles, a swage is often called an actual fitting. This is true to the extent that there is a "permaswage" fitting that is used onboard the orbiter. This is a specific type of swage fitting. The basic process of permaswage attachment involves locating two pieces of tubing into either side of a sleeve or ferrule and then placing the assembly into a collet. Pressure is applied to the collet, which presses the sleeve (or ferrule) into the tubing and forms a series of ridges between the sleeve and tubing that provide the seal (see fig. 7.4-1). This joint is well suited for use in high pressure, noncorrosive environments, such as orbiter hydraulics.

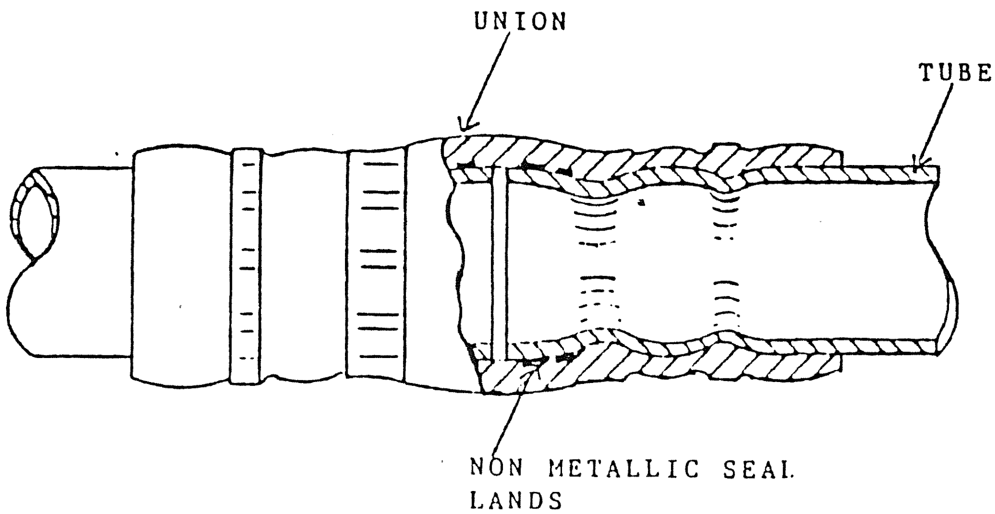


Figure 7.4-1.- Basic permaswage fitting.

7.4.1.2 Advantages/Disadvantages

a. Advantages:

- Comparatively lightweight
- Easy to install (can be done locally and in awkward positions with relatively little equipment)
- Relatively low cost joint
- Easy leak inspection that can be done visually (unlike welding that requires cumbersome x-ray methods)
- Minimal heat is applied to the joint during mating (therefore there is little reduction in material strength)

b. Disadvantages:

- The joint connection is permanent; it would have to be cut off to access line or install a new joint. In some instances a swaged joint can be disconnected provided that a sufficient amount of force is applied to break the joint.
- Care must be taken to ensure proper tube alignment in the sleeve/ferrule to prevent leakage.

7.4.2 Dynatube

7.4.2.1 Description

Dynatubes are one of the most common orbiter joints. There are basically two halves to this type of joint. One side has a threaded portion with a machined face. This half, the male half, is attached to tubing, etc. by one of three basic methods: welding, brazing, or swaging. The mating half, or female side, of this joint also has a highly machined and polished surface. It can be attached to the other side of tubing by any of the above methods. The two joint halves are simply screwed together to form a connection. A unique design feature of this joint is that it is self sealing under pressure by way of a groove that is cut into the inside of the joint (see figs. 7.4-2, 7.4-3, and 7.4-4). Orbiter hydraulic systems use swaging to attach dynatubes to their lines, the OMS/RCS plumbing primarily uses welding, and cryo and PCS use brazing.

7.4.2.2 Advantages/Disadvantages

a. Advantages:

- Generally easy installation/replacement, depending on the type of method used to attach the joint to the tube, etc. If swaging is used, it is an easy local operation. If welding or brazing is performed, it can be more difficult.

- Self sealing by way of a machined groove, forming a beam in the joint on which the fluid pushed.

b. Disadvantages:

- Must be handled with care during assembly/disassembly because the highly polished surface is easily scratched, leading to leakage

- Threads can be damaged easily, therefore care must be taken during assembly/disassembly

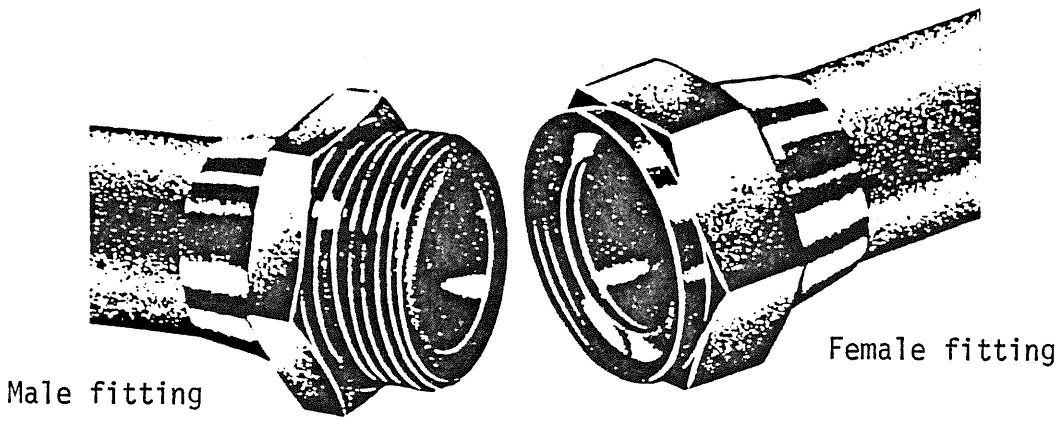


Figure 7.4-2.- Dynatube disassembled.

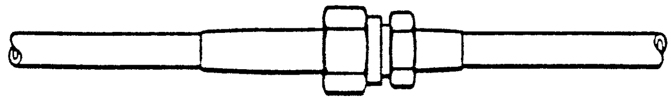


Figure 7.4-3.- Dynatube assembled and making a line connection.

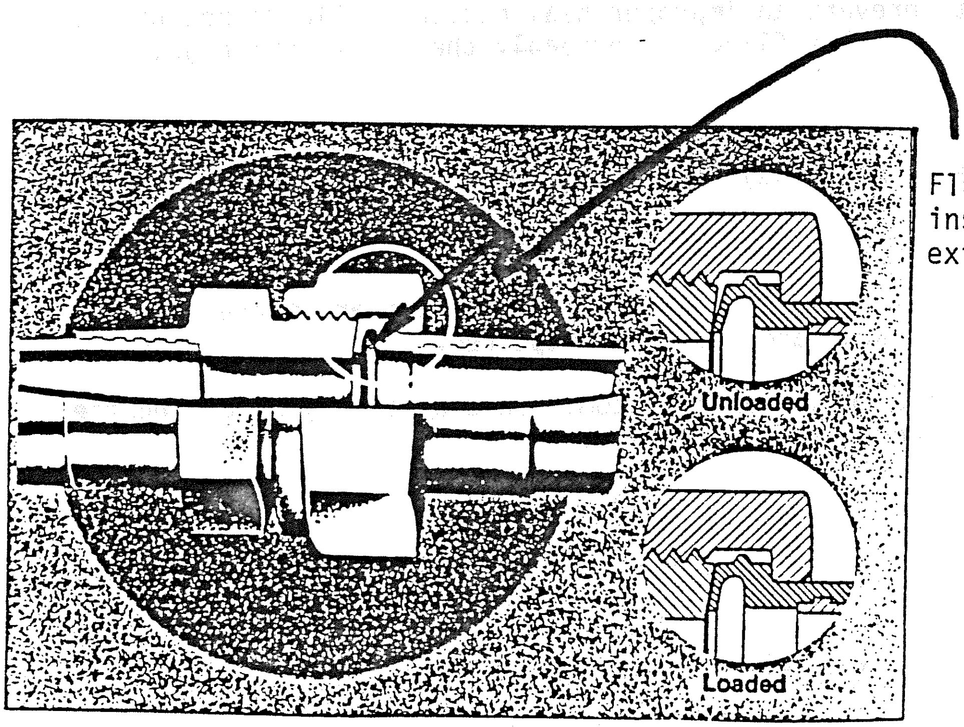


Figure 7.4-4.- Dynatube self seal.

7.4.3 Flanges

7.4.3.1 Description

There are many types of flange connectors. Since this type of joint is used mainly in the MPS area, the discussion is limited to two types of bolted flange connectors, the integral flange (an integral piece of the connector) and the separate flange (a ring with bolt holes, which rests on top of the connector). See fig. 7.4-5. The flange is usually welded to a line. Two opposing flanges are bolted together to complete a joint. A joint requires a seal between the two flanges to prevent line leakage. This seal is created using various gasket materials. Care must be taken when bolting flanges together to prevent an improper seal caused by flange bowing, faulty gasket alignment, etc. The flange is probably the least common joint aboard the orbiter.

7.4.3.2 Advantages/Disadvantages

a. Advantages:

- If the joint is damaged; i.e., threads on bolt are marred, the bolts are easily replaced and the joint repaired.
- Gasket seal changeout can be accomplished by simply unbolting the flange and changing out the gasket.

b. Disadvantages:

- It may be difficult in some cases (i.e., awkward flange locations) to access all flange bolts.
- It is typically a heavier joint.

7.4.4 Braze

7.4.4.1 Description

Brazing is used to make line-(tube-) to-connector joints in dynatubes and swage and flange-type applications. This process joins metals at temperatures somewhat below the temperatures required for welding (above 800° F). The basic difference between brazing and welding is that a joint is formed during brazing by wetting the base metal and connecting the two work pieces with the brazing material. Welding actually melts the base metal and combines with any filler metal to make a connection. The brazing metal is different than the base metal, whereas welding completely melts the base metal and becomes part of the joint. A brazed joint is semi-permanent while a weld is permanent.

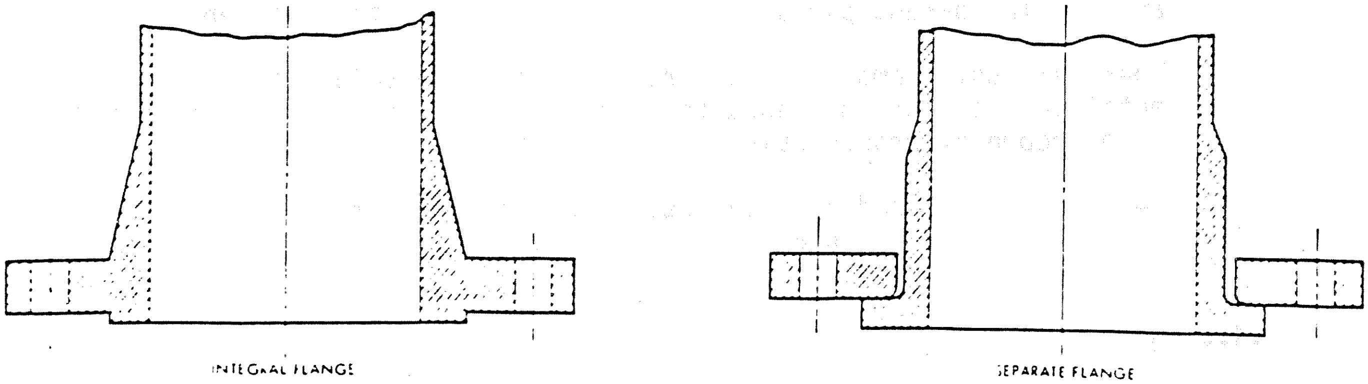


Figure 7.4-5.- Flanges.

7.4.4.2 Advantages/Disadvantages

a. Advantages:

- Most metals can be brazed by using a brazing metal that is compatible with the metals to be joined.
- Less heat is required for this operation, therefore local stress areas are less likely to form around joints. Welding requires much greater heat and thus increases local stress.
- This is a useful process in joining delicate parts.
- The process lends itself to mechanized operations.
- A joint can be reheated, broken apart, and rebrazed.

b. Disadvantages:

- Although a joint can be rebrazed, it is weakened in the process, thus it cannot be done very often on any one particular joint.
- It could be difficult to x-ray on location if the fitting is in an awkward location.

- Inadvertent joint reheating may soften braze and the joint could come apart. The parent parts would be unaffected if this happened.

- There are some compatibility problems with the selection of braze metal and its bonding characteristics with the parent metal; this can be overcome by proper selection of the braze metal.

- X-raying the brazed area can become an involved process if it is performed locally (on a completed orbiter or one on the pad).

7.4.5 Weld

7.4.5.1 Description

Welding is used to make line-(tube-)to-connector joints in dynatube and swage and flange-type applications. This process joins metals at temperatures well above those used in brazing. The basic difference between welding and brazing is that a joint is formed during welding by actually melting the base metal and combining it with any filler metal to make the connection, while brazing wets the base metal and connects the two work pieces with the brazing material. A weld is permanent while brazing is semi-permanent.

There are many different forms of welding. One of the more common orbiter welding methods is inert gas welding. The process basically consists of a wire fed system with a current flowing through it which is melted to form a joint. It uses an inert gas such as argon or helium that flows over the wire and joint during the melting of the metal and produces a weld that is largely free of atmospheric contaminants. There are several types of weld joints. A common one used throughout the orbiter is the sleeve joint. These are designed such that either end of a line is placed into a sleeve and welded together (see fig. 7.4-6). This type of weld is prevalent in the OMS/RCS plumbing. Equipment required to make these welds ranges from hand held to semiautomatic to fully automated systems.

7.4.5.2 Advantages/Disadvantages

a. Advantages:

- Inert gas welding can be performed in awkward positions, in some cases joints can be welded in place on the vehicle.

- A very strong, virtually leak free joint is created.

- Inert gas provides a fairly clean weld (free of contaminants that could lead to corrosion or a weaker joint).

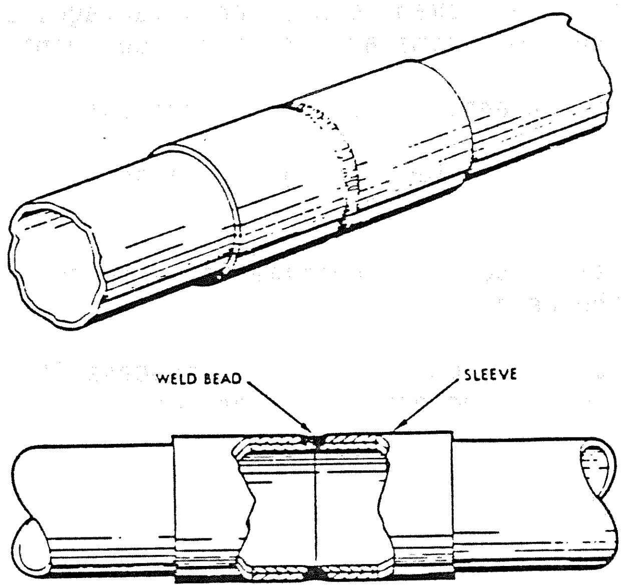


Figure 7.4-6.- Welded sleeve joint.

- The equipment for some forms of welding is highly portable.
- Some types of welding lend themselves to mechanized operations.

b. Disadvantages:

- Although portable equipment can be an advantage, some types of welding require cumbersome and/or fixed equipment.
- The joint is permanent, so it must be cut out to remove it.
- It may be difficult to keep the joint clean during the welding operation.
- Care must be taken to avoid detrimental effects of heat on the metal surrounding the weld.
- X-raying welds can become an involved process if it is performed locally (on a completed orbiter or one on the pad).

7.4.6 Quick Disconnect

7.4.6.1 Description

Quick disconnects (QD's) cover a wide variety of applications. These are simply fittings that are self sealing when disconnected. QD's are used anywhere from an MPS 17-inch valve to the galley water supply lines in ECLSS.

Discussion of this type of joint is limited because of the many different types and applications of QD's. To obtain any further information on a particular type of joint in a given subsystem, consult a Rockwell Component Handbook, Specifications Book, or subsystem manager.

7.4.7 References

1. Aerospace Fluid Component Designers Handbook, Vol. I, Rev D, 1970, pp. 5.12.1-1 to 5.12.3-1, 5.12.8-1.
2. E. Paul DeGarmo: Materials and Processes in Manufacturing, Macmillan Publishing Co., Inc. 1979, pp. 17, 831, 832, 838-841, 909 and 910.
3. Resistoflex, Resistoflex Corporation, 1971. (This is a small handbook that can be attained through reference no. 4.)
4. J. Wiltz, E & D, Engineering Directorate, Propulsion and Power Division, Systems Branch/EP2

7.5 ELECTRON BEAM WELDING

7.5.1 Introduction

Many components used in the SSME have electron beam welds. A example of one such component is the high pressure fuel turbopump (HPFP) turbine first stage forward platform seal. This seal is manufactured as an INCONEL 625 ring which is electron-beam welded to Rene 41. After a description of electron beam welding, this systems brief will expand upon one problem encountered with electron beam welding on the HPFP turbine first-stage forward platform seal.

Electron beam welding is suitable for joining almost all metals, similar and dissimilar, and can often be done without using filler material. The most outstanding characteristic of electron beam welding is its very low total heat input to the materials being welded. The low total heat input is reflected in low distortion of the materials (such as shrinkage) and low metallurgical disturbance. In addition, deep penetration welds having very narrow fusion zones can be obtained. Because the process is usually done in a vacuum, the oxidation effects are eliminated.

One of the disadvantages is that there are higher residual stresses as the result of the rapid solidification and cooling of liquid metals; however, this can be minimized by stress- relieving the part (i.e., hold the temperature high longer). Residual stresses can be a factor for cracking. Another disadvantage is that porosity of the final weld is possible due to the high temperature at the location of the weld and to the low pressure environment. Any gases generated by impurities have a harder time escaping since the fusion zone is narrow and the process is done under a vacuum. Quality control of the raw materials helps minimize this concern. Another factor of good welding is the manipulation of the workpiece being welded. The joint must be moved at a constant speed and remain directly beneath the beam within tight tolerances since the weld is narrow. Otherwise, the beam will mistrack and there could be lack of fusion or insufficient penetration.

7.5.2 Electron Beam Welding Principle

Figure 7.5-1 shows a simplified diagram of a typical electron gun used for electron beam welding. The following describes how the electron gun works:

a. The filament is often a simple filament (usually tungsten) and is heated by passing a current through it.

b. By heating the filament, it emits a cloud of electrons.

c. The specially-shaped cathode cup imposes an electrostatic force on the cloud and shapes it into a beam.

d. A high voltage (which can be 15 to 150kV) is applied between the cathode and an anode.

e. The resulting potential will cause the electrons to accelerate, imparting kinetic energy to the electrons.

f. The anode has a central hole to pass the beam.

g. The electrons coming through the anode will have a low power density (i.e., the number of electrons per cross section of the beam will be small).

h. The energy in the beam can be concentrated by focussing the beam with a magnetic lens. Therefore, very high power densities can be achieved at the focal point (where the welding takes place). The focal point and power density of the beam are set by controlling the current through the lens.

i. When the beam impacts solid material, electrons penetrate into the metal.

j. Heat is generated due to the conversion of kinetic energy of the electrons. (Note that X-rays are released during the electron bombardment process; therefore, adequate shielding is required.)

k. This heat is so intense that almost instantaneous melting and vaporization takes place.

l. The fusion of the two materials forms the weld.

m. A very deep, narrow weld can be produced.

n. To keep the cathode assembly from oxidizing and to extend the life of the filament, the cathode assembly must be operated in a vacuum environment of the order of 10 to the minus 4 torr (1 torr = 1 atmosphere/760).

o. The welding chamber can operate at a vacuum of the order of 10 to the minus 1 or 2 torr without causing unacceptable beam scatter.

7.5.3 Mechanism of Penetration of Metal By Electron Beam

Refer to figure 7.5-2.

a. The electron impacts the metal.

b. Vapor and liquid is produced.

c. The vapor has a lower density than the solid; therefore, the beam can pass through it and strike and vaporize the material at the bottom of the vapor cloud. This operation is progressive.

d. With the vapor/liquid hole, a weld can be made by translating the hole along a predetermined joint line.

e. The material behind the advancing hole solidifies as the heat source is removed.

f. The total time required to produce the weld is a function of material thickness, energy available in the beam, and thermal conductivity of the material.

7.5.4 Quality and Process Control

a. Raw material inspection and preparation

(1) Inclusion content of raw material is critical (porosity concern)

(2) Edge preparation

(a) Mating surface

- Chemically cleaned (affects infusion)
- Interference fit of pieces being welded (affects infusion)
- Should be parallel to electron beam (affects penetration)

b. Welding process

(1) Vacuum to eliminate scattering of beam and oxidization

(2) Guidance
if not horizontally aligned at point of beam impingement, could result in mismatch

(3) Electron gun optics important

(4) Trained people

c. Weld inspection

(1) Visually to detect gross external defects

(a) Pores

(b) Mismatch

(c) Partial penetration

(d) Underfill

(2) Internal inspections

(a) Ultrasonics

(b) Dye penetrant

(c) X-ray (however to detect defect requires almost perfect x-ray beam orientation to the defect)

7.5.5 HPFP Turbine First Stage Forward Platform Seal Weld Problem

In January 1988, analysis of a certification HPFP which had accumulated approximately 7600 seconds of testing showed that a weld on a seal was mistracked (due to weld head misalignment), resulting in only a 15 percent weld penetration. That weld was on the first stage forward platform seal, which is often referred to as a fishmouth seal. Figure 7.5-3 shows the location of the seal in the turbine end of the turbopump. Figure 7.5-4 is an enlargement of the seal. As can be seen, the platform of the blade fits into the "fishmouth." This is done to minimize bypass leakage. Obviously, the clearance between the blades and the seal must be controlled to reduce bypass leakage yet minimize rubbing. Light blade platform rubbing on the upper and lower lip of the fishmouth seal can occur due to the controlled clearances required to maintain turbine efficiency. The rubbing is monitored by post-test/flight borescope inspections and controlled by field inspection.

7.5.6 Seal Characteristics

Prior to 1982, the seal was made entirely of Rene 41. However, due to cracking problems, the seal was redesigned to a bimetallic configuration (figure 7.5-4). This redesign resulted in a significant reduction in cracking. The inner structure is made of Rene 41 which has good rupture strength. The active platform seal portion is manufactured as an INCONEL 625 ring which is electron-beam welded to the Rene 41. An electron beam weld is a clean weld with a small heat affected zone and therefore results in minimal distortion. INCONEL 625 was chosen because of its thermal properties, high cycle fatigue life, and insensitivity to hydrogen environment embrittlement (HEE) at operating temperatures. The seal is secured to the turbine support by 24 A-286 corrosion resistent (CRES) bolts and 321 CRES cupwashers. A static seal, shown in figure 7.5-4, is located between the platform seal and the turbine support to prevent direct exposure of the bolts to the high temperature turbine gases.

7.5.7 Seal Criticality

The first stage forward platform has criticality 1 and 1R failure modes. The criticality 1R failure mode is seal leakage. To better understand the effects of seal leakage, see figure 7.5-5. Figure 7.5-5 shows the hot gas (hydrogen rich steam) flow path from the fuel preburner (FPB) through the turbine and out the turnaround duct. Flow from the turnaround duct is discharged into the hot gas manifold. The hot gas manifold conducts the hot gas to the main injection elements of the main chamber injector to be mixed and burned with liquid oxidizer. If the seal leaks excessively, the hot gas will be diverted from the turnaround duct flow path. Less hot gas would be output from the turnaround duct. Therefore, the power output by the turbine would be reduced, resulting in reduced pump speed, flow, and discharge pressure. The controller would open the fuel preburner oxidizer valve (FPOV) to increase the FPB oxidizer flow. The increased flow to the FPB would increase the HPFT turbine discharge temperature. If the temperature exceeds the redline, engine shutdown would occur. Groundrules for the critical items list (CIL) require that any premature shutdown be considered a criticality 1R because an abort could result from the shutdown.

Seal leakage also has a secondary effect. The effectiveness of the cooling circuit would be diminished. In figure 7.5-6, the cooling circuit is shown. Excessive leakage would allow hot gas to enter the coolant circuit. Coolant for components such as blades and bearings is important.

The criticality 1 failure mode is excessive rubbing of the seal which could lead to structural failure of the turbine blades. Failure of the blades results in immediate loss of turbine power and rotor unbalance. Extensive turbine damage would result from impact and excessive temperatures. The pump would stall or seize and stop the fuel flow. The resulting pressure surge could burst the pump inlet duct. The engine would go LOX rich, and uncontained engine damage would result.

7.5.8 Impact

Visual inspections done on 56 seals showed 38 seals with mistracked welds and incomplete penetration. The seal with the 15 percent weld penetration was the worst case identified. A fracture mechanics analysis concluded that the minimum acceptable penetration is 50 percent.

All of the pumps planned for use on STS-26 had seals from the same lot as the seal that had the bad weld. Because the welds can not be verified while the seal is in the pump, the decision was made to return all delivered flight pumps for seal replacement and regreen run.

7.5.9 References

M. J. Fletcher, Electron Beam Welding, 1971.

M. M. Schwartz, ed., Source Book on Electron Beam and Laser Welding, American Society for Metals, 1981.

R. G. Taylor, "NDT of Electron Beam Welded Joints" AGARD conference proceedings no. 398, Advanced Joining of Aerospace Metallic Materials, 1987.

E. Kearney, D. White, and W. Ricci, "Weld with Monitor and Control System," AGARD conference proceedings no. 398, Advanced Joining of Aerospace Metallic Materials, 1987.

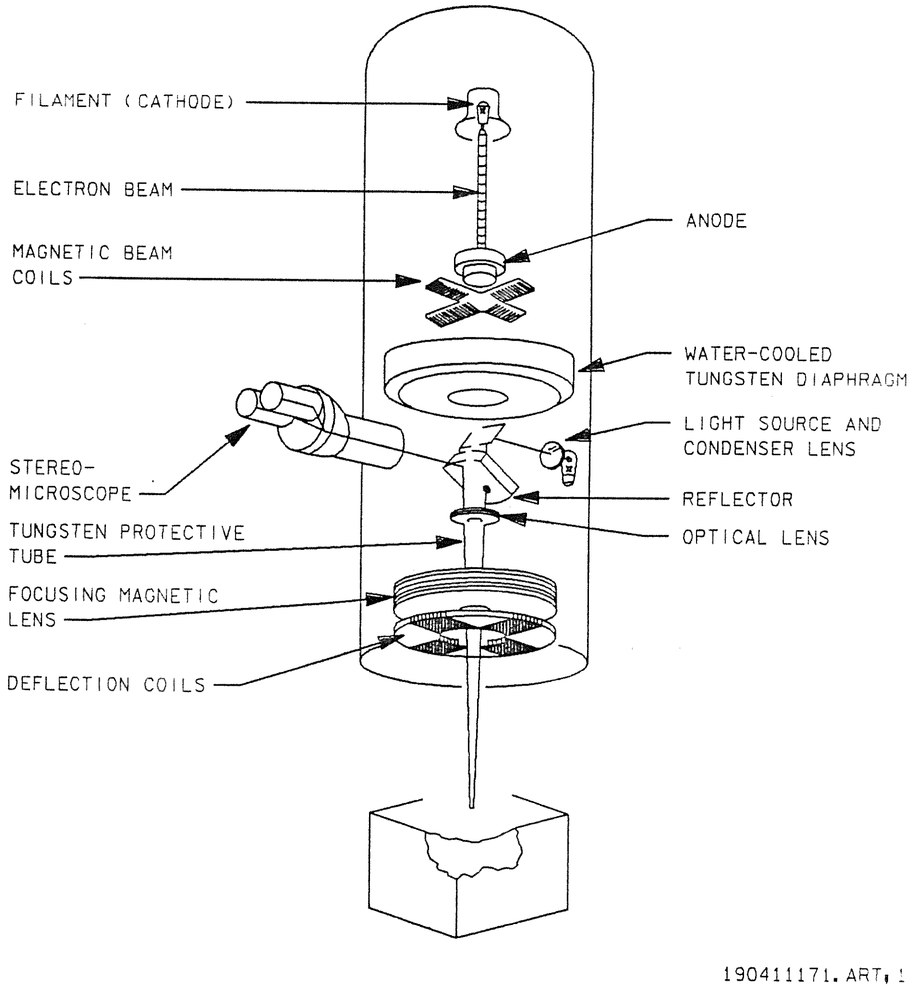


Figure 7.5-1.- Diagram of electron gun used for electron beam welding.

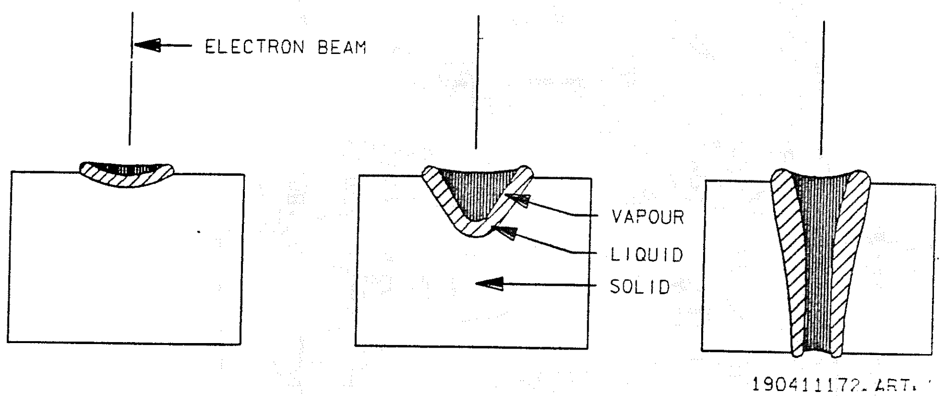


Figure 7.5-2.- The mechanism of penetration of metal by an electon beam.

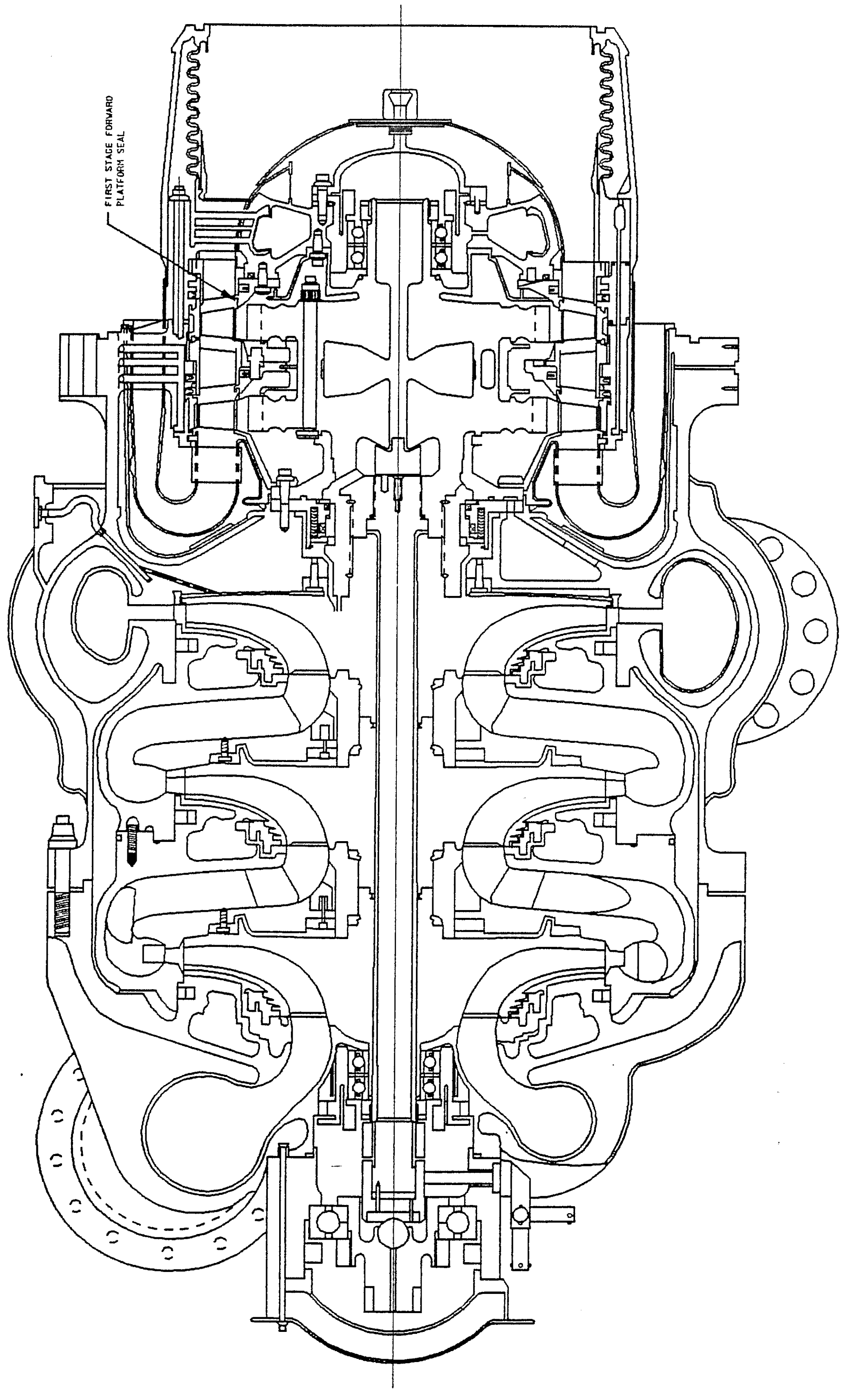


Fig. 7.5-3 - Location of seal in high pressure fuel turbopump.

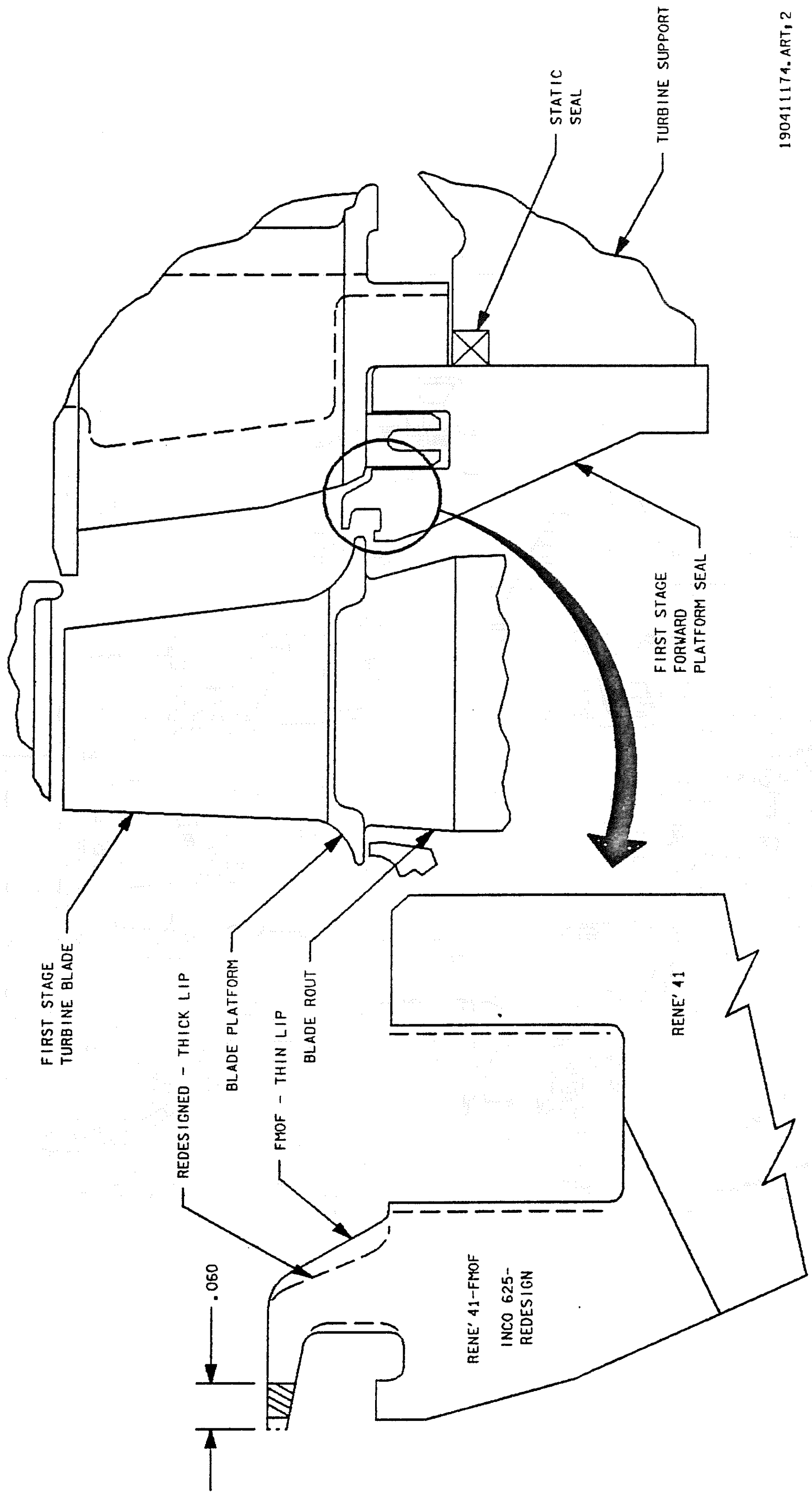


Figure 7.5-4.- HPFTP fishmouth seal configuration.

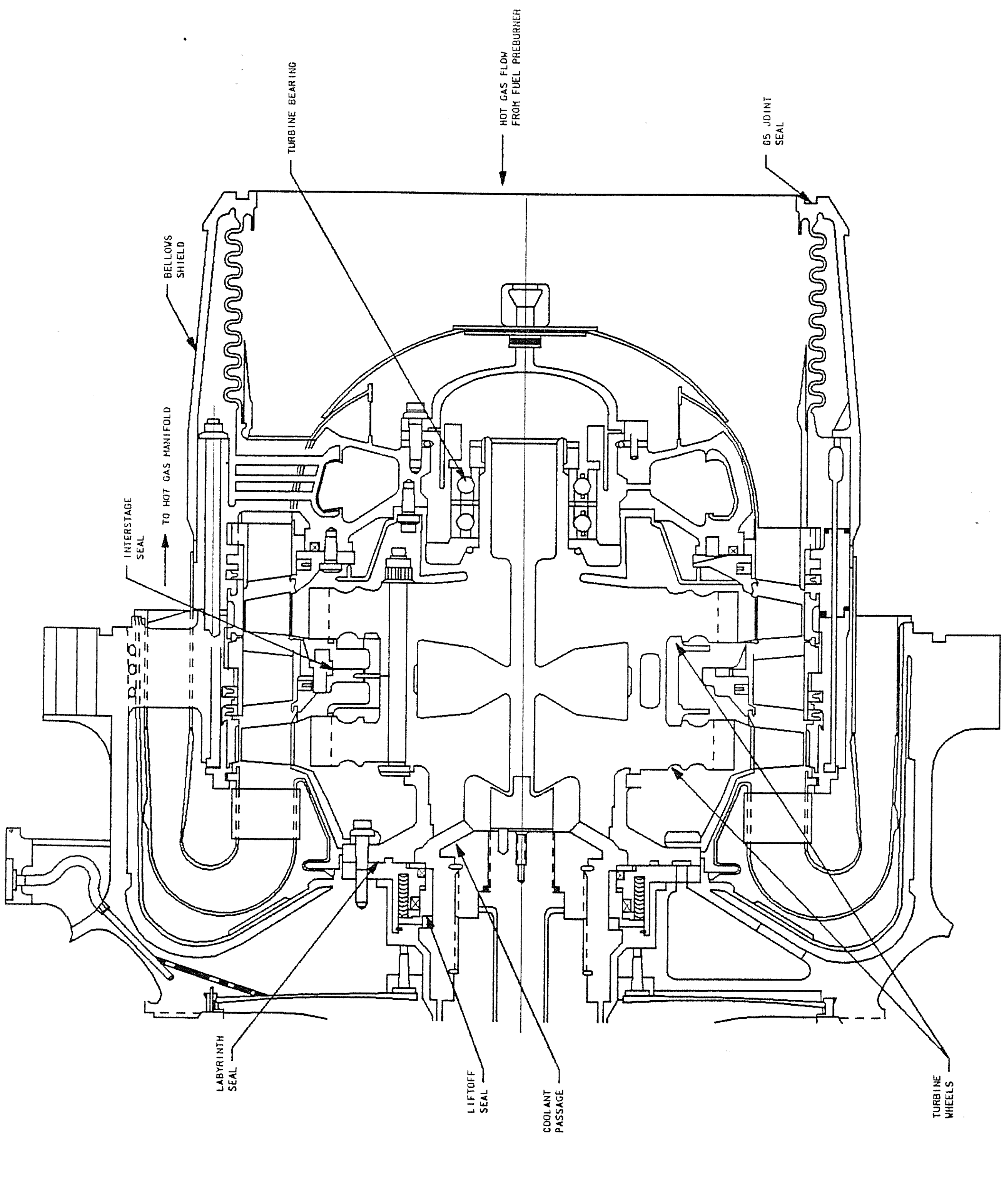


Figure 7.5-5 .- Hot gas flow through turbine.

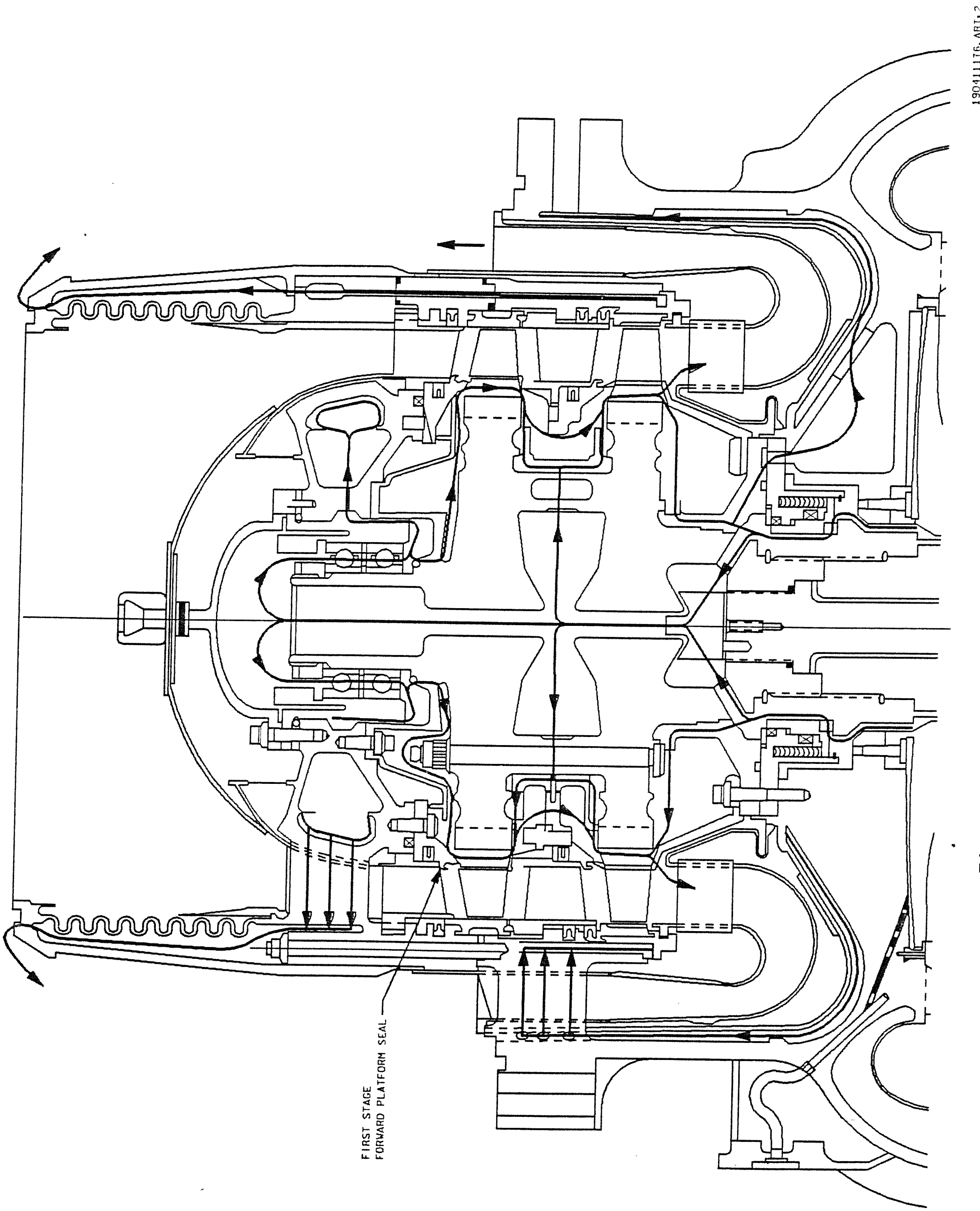


Figure 7.5-6 - Turbine coolant circuit.

7.6 PYROTECHNIC INITIATOR CONTROLLER

7.6.1 Introduction

The pyrotechnic initiator controller (PIC) provides the energy impulse necessary to discharge a NASA standard initiator (NSI), type 1 (NSI-1). The PIC supplies 100 millijoules minimum to the initiator for 5 ms from electrical energy stored in an internal capacitor bank.

PIC circuits include both operational and test circuits as follows:

Operational

- ARM
- FIRE 1
- FIRE 2
- Capacitor bank voltage analog

Test

- Resistance test
- Load test

7.6.2 Overview

The PIC requires three input commands to provide the pyro output energy pulse to the NSI-1. These are the ARM, FIRE 1, and FIRE 2 commands. The PIC will not fire if the ARM and FIRE commands are applied out of sequence. The ARM command provides voltage to the energy storage capacitor bank. The FIRE 1 command provides a switch closure in the negative return output leg, and the FIRE 2 command provides a switch closure in the positive output leg of the NSI-1 firing circuit.

In addition to these inputs, there are two pyro circuit status checking test circuits, the resistance test circuit and the load test circuit. The resistance test checks prefiring circuit continuity and resistance, resulting in a GO/NO-GO discrete signal output. The load test circuit provides post-firing verification of the PIC output energy level, generating a GO signal if the energy level is within designated limits.

See figure 7.6-1.

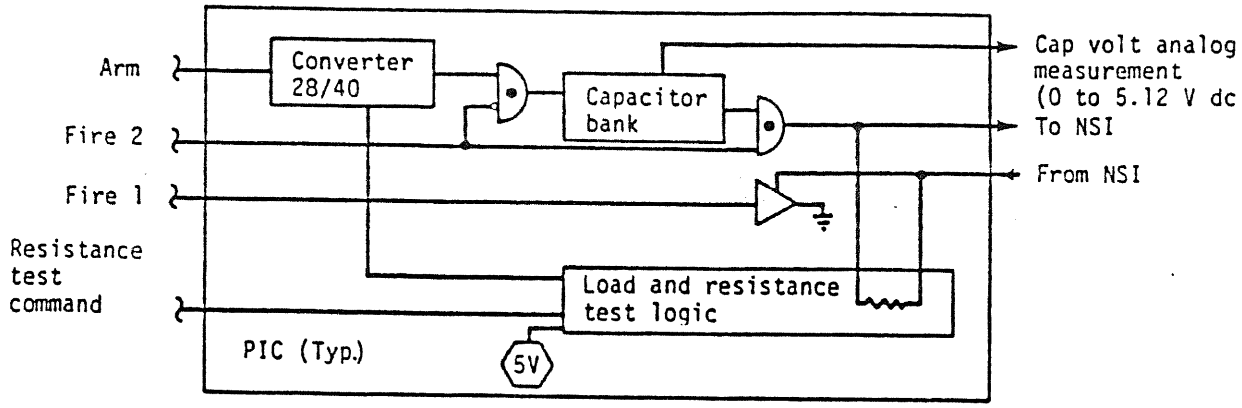


Figure 7.6-1.- Pyrotechnic initiator controller circuit functional diagram.

7.6.3 Operation

7.6.3.1 ARM Command

The ARM power supply circuit generates three separate dc power levels when positive voltage and return is applied to its input. The ARM command input voltage is 28 V dc nominal with a minimum of 20 V dc and a maximum of 32 V dc. Rise time under these conditions is less than 10 milliseconds.[a] The ARM signal must be applied to the PIC for a minimum of 0.25-second before the FIRE signals. The ARM command has a duration of 1 second minimum and must be maintained for at least 5 milliseconds after the issuance of both FIRE commands. The ARM current has a 0.5-ampere maximum value, decaying to 0.2 ampere in 1 second. Filter inrush is 30 millijoules maximum within 10 milliseconds. Following the issuance of the ARM command or the FIRE 2 command, a 300-milliampere transient of 10-milliseconds maximum duration may occur over and above the 0.5-ampere maximum.

The dc power levels generated as a result of the ARM command are 5 V (±10 percent) and 12.5 V (±10 percent) for the power input of the load test circuit, and 41.1 V dc (±10 percent) as the charging voltage for the capacitor bank.

The response time for supplying these voltages is within 30 milliseconds after the ARM command is received. Should it be desirable to remove the ARM command with no FIRE command present, the capacitor bank voltage will discharge to a maximum of 1.5 V dc in less than 30 seconds as indicated by the capacitor voltage analog signal. The maximum voltage level of the ARM command in the OFF state is 1.7 V dc.

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[a]Preliminary testing indicates that the PIC capacitor bank will continue to charge with an input voltage less than 20 V dc. Approximate values show that for a low input voltage of ~13 V dc, the capacitor bank will still charge up to ~37 V dc. Rise time is ~10+ milliseconds.

7.6.3.2 FIRE 1 and FIRE 2 Commands

The FIRE 1 command must precede or be concurrent with the FIRE 2 command in order to cause the PIC capacitor bank to discharge through the pyrotechnic initiator. The FIRE 2 command must not be initiated less than 1 second after the ARM command. The voltage requirements for the FIRE 1 and FIRE 2 commands are the same as those for the ARM command. The FIRE 1 and FIRE 2 currents have a maximum of 10 milliamperes (each). Both commands must be provided simultaneously for a minimum of 10 milliseconds to effect a discharge of the PIC capacitor bank through the pyrotechnic initiator. The voltage applied to the FIRE command circuits in the OFF condition has a value of 1.7 V dc maximum.

7.6.3.3 System Test Circuits

System test circuits are provided to ensure that the live pyro resistance after NSI-1 installation is within allowable limits (resistance test) and that the PIC output energy level is sufficient to discharge the pyro (load test).

The load test circuit is triggered each time a PIC is discharged. This test is performed during ground test operations. For purposes of this test, a dummy load is inserted instead of the NSI-1. This dummy load consists of a 0.931-ohm (±1 percent) wire wound resistor in series with two diodes. The load test is performed by generating an ARM command followed by both FIRE commands not less than 1 second later. If the PIC pulse generating circuitry is operating normally, a GO signal is generated within 10 milliseconds from the second FIRE command.

A graph of the voltage decay versus the time curve of the capacitor bank is provided on figure 7.6-2. The load test circuit voltage must decay at a specific rate in order to satisfy the requirements for a GO condition. As indicated by the graph, the output voltage starts out at approximately 40 V; at 0.6 milliseconds the level must pass through the "window" as shown. If these requirements are met, the load test circuit will indicate a GO condition.

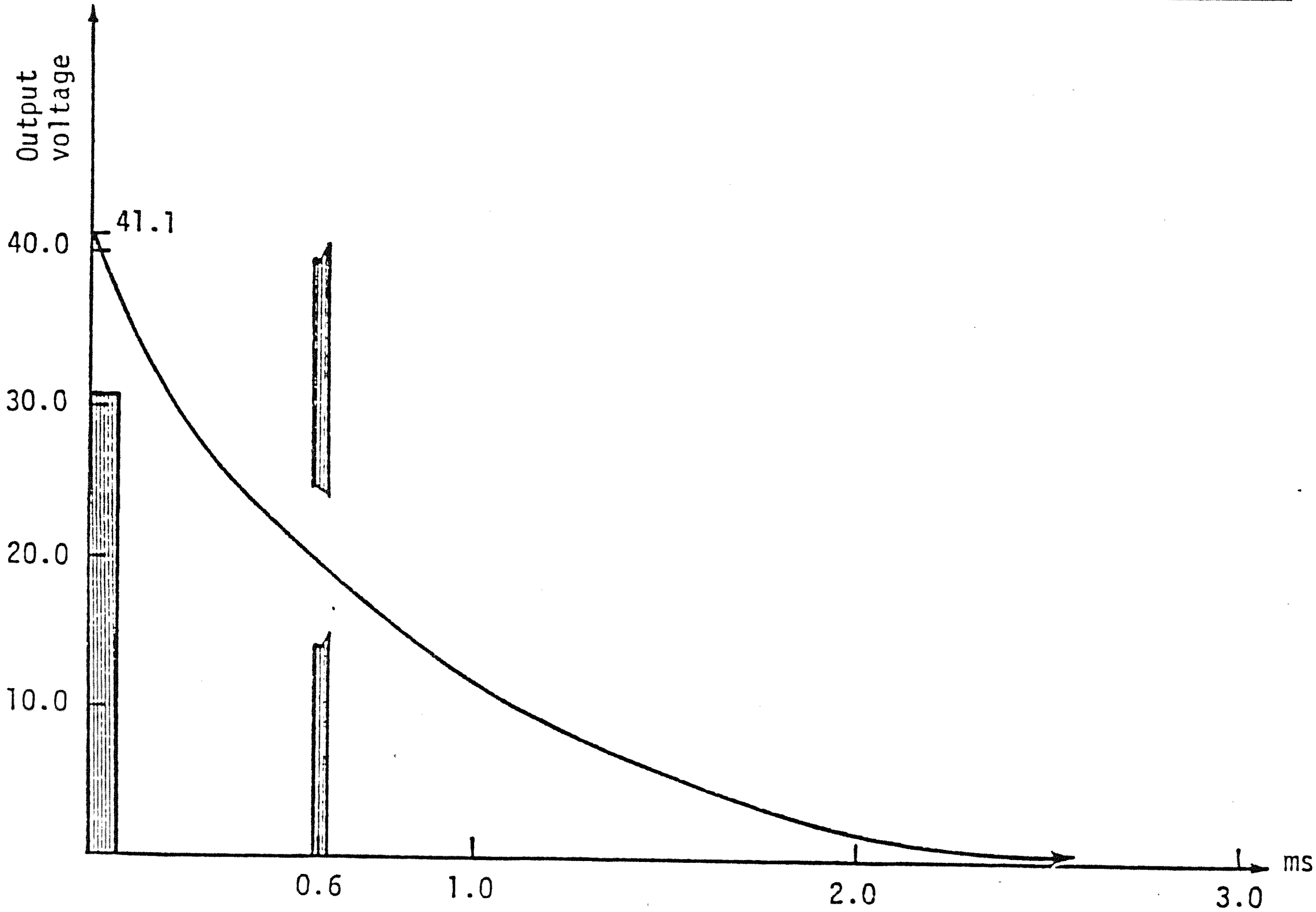

Figure 7.6-2.- Load test PIC circuit voltage decay.

The resistance command voltage has the same requirements as the ARM command voltage. The resistance command current to the PIC has a maximum value of 70 milliamperes. Upon receipt of the resistance test command, the PIC performs a resistance test of the circuit connected to the PIC pyro output terminals. This PIC resistance test is performed to provide instantaneous verification before flight that the dummy load has been removed and the NSI installed with all the interface connections between the PIC and the NSI properly mated. This test also performs a check of wire/cable integrity and NSI electrical integrity.

Receipt of the resistance test command (28 V dc nominal) energizes the PIC circuit, providing a maximum of 1 V dc and a maximum current of 20 milliamperes at the pyro output terminals. The external resistance of the circuit must be greater than 1 to 1.25 ohms and less than 1.75 to 2.1 ohms to generate a GO signal. If the dummy load has not been removed before this test, the two diodes in the load will guarantee a NO-GO response. The resistance test discrete initializes a NO-GO state. Removal of the resistance test command will also result in a NO-GO signal (fig. 7.6-3).

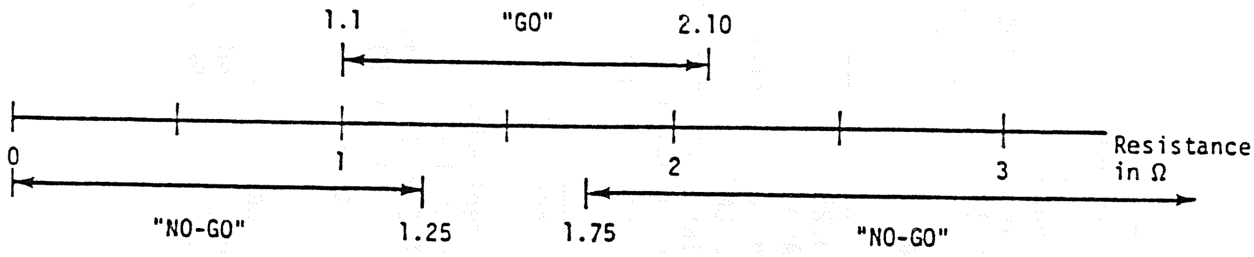

Figure 7.6-3.- Resistance test command.

7.6.3.4 PIC Outputs

The PIC outputs include two discrete signal outputs, the resistance test discrete signal and the load test discrete signal. Each of these discrete outputs consists of an isolated switch closure, with a closed switch corresponding to a GO state and an open switch corresponding to a NO-GO state. The positive side of a 5-V dc or a 12-V dc (±10 percent) power supply is provided continuously to the PIC for connection only to each of these two switches. The maximum current through each of these switches during the closed (GO) state is 0.5 milliamperes. A 1200-ohm resistor is located in series with each of these switches. The maximum voltage drop through a closed switch is 0.5 V. Maximum leakage through an open switch is 50 amperes.

The load test discrete signal output provides an indication that the energy level available from the PIC during ground test operations is within the range limits for detonation of the NSI. This test is conducted as described above. The FIRE circuit can be expected to deliver 75 millijoules[a] of energy to an NSI-1 of 0.95 to 1.15 ohms; however, the test will not reject a PIC operating within the entire PIC operational temperature range. An open circuit condition in the PIC or in the circuit containing the NSI-1 will result in a NO-GO output within 20 milliseconds after the ARM command has been generated. Removal of the ARM command will result in a reset of the load test discrete signal to a NO-GO state.

The resistance test discrete signal output (generated as a result of the resistance test) is present within 10 milliseconds of the issuance of a resistance test command. The resistance test discrete signal output results in a GO/NO-GO, based on the criteria listed in tables 7.6-I and 7.6-II.

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[a]The 75-millijoule energy level is the known minimum requirement for proper NSI-1 firing. One hundred millijoules for 5 milliseconds is set as the minimum energy level required, allowing for a 25-millijoule safety margin. Analytically, however, the worst case minimum output from the capacitor band is 106.9 millijoules, which again exceeds the 100-millijoule requirement. In practice, output levels are typically around 200 millijoules in the ambient environment, allowing for a significant system operating margin.

TABLE 7.6-I.- PYROTECHNIC INITIATOR CONTROLLER INPUTS

	Purpose	Voltage	Current	Duration/sequencing
ARM CMD	Provides arming for capacitor bank	28 V dc (nom.) 20 (min.) to 32 (max.) V dc 1.7 V dc (max.) in OFF state	0.5 A (max.) decaying to 0.2 A in 1 sec	1 sec (min.) duration applied at least 0.25 sec before FIRE 1 and maintained at least 5 msec after issuance of both FIRE CMD's
FIRE 1 CMD	Req'd coincident with FIRE 2 CMD to cause PIC capacitor bank to discharge through NSI	28 V dc (nom.) 20 (min.) to 32 (max.) V dc 1.7 V dc (max.) in OFF state	10 mA (max.)	Issued not <0.25 sec following ARM CMD Issued before or concurrent with FIRE 2 CMD FIRE 1 and FIRE 2 CMD's must overlap for at least 10 msec to effect firing of NSI
FIRE 2 CMD	Req'd coincident with FIRE 1 CMD to cause PIC capacitor bank to discharge through NSI	28 V dc (nom.) 20 (min.) to 32 (max.) V dc 1.7 V dc (max.) in OFF state	10 mA (max.)	Issued not <1 sec after ARM CMD Issued concurrent with or following FIRE 1 CMD FIRE 1 and FIRE 2 CMD's must overlap for at least 10 msec to effect firing of NSI
Resistance test	Req'd to perform resistance test of the ckt. connected to PIC pyro output terminals (i.e., verification before flight that dummy load has been removed and NSI installed and properly mated)	28 V dc (nom.) 20 (min.) to 32 (max.) V dc 1.7 V dc (max.) in OFF state	70 mA (max.)	Duration 1.1 sec (min.). CMD shall not be issued simultaneously with ARM or FIRE CMD's

TABLE 7.6-II.- PYROTECHNIC INITIATOR CONTROLLER INPUTS

	Purpose	Characteristics
Capacitor voltage analog signal	Provides a continuous measurement of the PIC capacitor bank voltage	Output range is 0 to 5.12 V dc (nom.) corresponding to 0 to 40.75 V dc (max.) capacitor bank voltage
Resistance test discrete signal	Provides GO or NO-GO indication for the resistance test performed on external ckt. connected to PIC initiator firing pulse terminals	Resistance test discrete signal output shall be present in <10 msec after issuance of resistance test command External resistance ranges: <1.25 Ω = NO-GO >1.25 Ω and <1.75 Ω = GO >1.75 Ω = NO-GO
Load test discrete signal output	Provides indication that the energy for a detonation of NSI is available	Performed during ground test operations with dummy load inserted Load test is performed by generating the ARM CMD followed by both FIRE CMD's not <1 sec later; a GO condition shall be generated within 10 msec from FIRE 2 CMD 100 mJ for 5 msec is set as the min. energy level req'd A NO-GO condition will be indicated within 20 msec after the ARM CMD has been generated; removal of ARM CMD will reset load test discrete signal to NO-GO state

Initialization of the circuit results in a NO-GO. Removal of the resistance test command causes the resistance test discrete signal output to be reset to a NO-GO state.

The initiator firing pulse will be generated in response to the ARM command followed by both FIRE commands. It provides a capacitive discharge pulse, the energy of which at the NSI-1 interface has a minimum value of 40 amperes. In the armed state, without both FIRE commands, the leakage to the NSI has a maximum value of 10 milliamperes.

The capacitive voltage analog signal provides a continuous measurement of the voltage on the PIC capacitor bank. The output range of the capacitor voltage analog signal is 0 to 5.12 V dc nominal, which correlates to 0 to maximum (40.75 V) capacitor bank voltage.

The application of the command and test signals required for PIC operation is represented in the timeline of the PIC ON/OFF cycle (fig. 7.6-4).

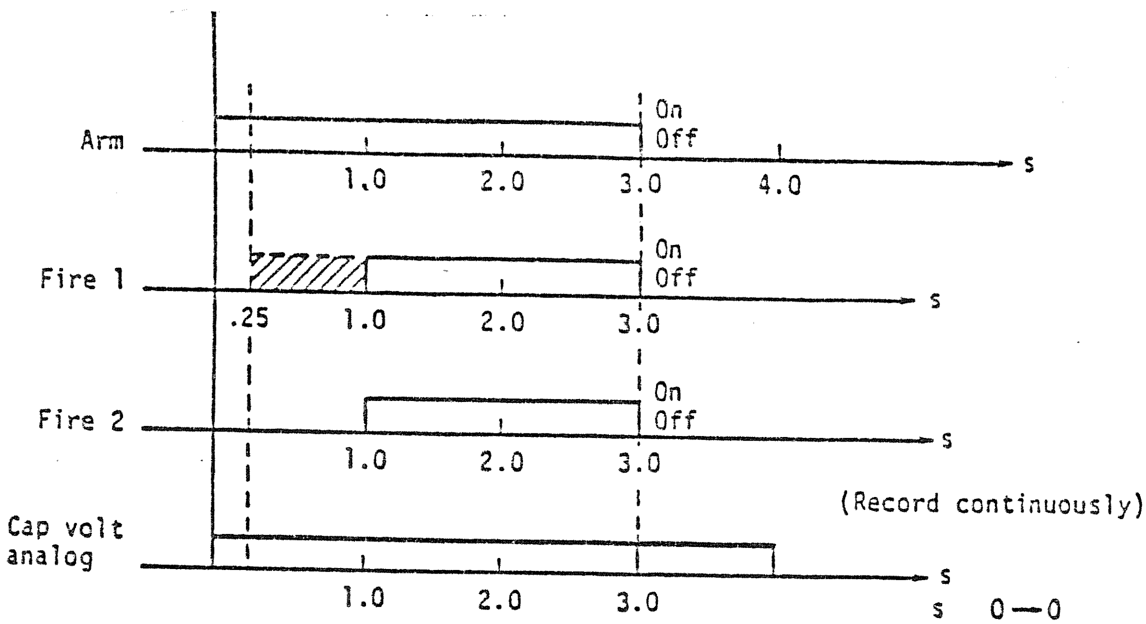

Figure 7.6-4.- The PIC command/test signals.

7.6.4 PIC Locations and Monitoring

A compilation of the PIC locations in the orbiter, the external tank (ET), and the solid rocket boosters (SRB's) is included under table 7.6-III of this brief. This table indicates PIC location, safing, and the Space Shuttle Systems Handbook (SSSH) drawing number in which the PIC appears.

The PIC's are monitored by EECOM on several different manual select keyboards (MSK's). The PIC capacitor voltage is the parameter most frequently monitored, although some MSK's are used to indicate whether or not ARM and FIRE commands have been sent. MSK's 0262 and 0603 display the capacitor voltage for the fire suppression bottles in avionics bays 1, 2, and 3. They also display the capacitor voltages on the PIC's in the landing gear pyros, including extend actuator nose landing gear (NLG), emergency extend NLG, left main gear (LMG), and right main gear (RMG), systems A and B. Parameter numbers, limits, and corresponding display decoder drive (DDD) format numbers for the PIC's monitored by EECOM are included in table 7.6-IV. Referencing this table shows the limits for the fire suppression bottle capacitor volts to be set at -0.1 to +0.2. These limits are kept low so that an event light will light at the first sign of capacitor charging. The low limit of -0.1 volts is chosen as a means of preventing an event light from coming on for a capacitive charge of zero.

All landing gear (LG) pyros, except those for the NLG extend actuator assist, are armed when the LG ARM pushbutton indicator (pbi) is pushed. The NLG extend actuator assist pyro is armed 2 seconds after the LG DOWN pbi is pushed. Only the extend actuator NLG PIC's, however, will discharge through an NSI during normal LG deployment. The extend actuator NLG capacitor volts will drop back to zero instantaneously as the PIC discharges. The emergency extend NLG, LMG, and RMG capacitor volts in this case will bleed back down to zero in 2 to 3 seconds. Should it be necessary to fire the emergency extend PIC's, the emergency extend capacitor volts will drop back to zero instantaneously. A summary of the fire suppression bottle and landing gear PIC capacitor volts is kept on the appropriate history tab MSK's (table 7.6-IV).

The remaining PIC's monitored on MSK 0602 are the range safety system and the ET/SRB separation PIC's. The 0602 MSK displays the PIC capacitor volts, ARM, FIRE, and SAFE and ARM commands. These parameters also appear on history tabs and have associated event lights. The parameter numbers, limits, MSK numbers, and DDD formats for the range safety system and the ET/SRB separation also appear in table 7.6-IV.

TABLE 7.6-III.- PIC LOCATIONS

PIC	Location		How safed		SSSH dwg. no.
	Sys. A	Sys. B	Sys. A	Sys. B	
FLC PIC functions					
Orb./ET fwd. sep.	FLC1[a]	FLC2[b]	MEC	MEC	5.3
Av bay fire extgh.					
Av bay 1	FLC2	--	015:C fire suppr. bay 1 - open	--	5.1
Av bay 2	FLC3	--	016:C fire suppr. bay 2 - open	--	5.1
Av bay 3A	FLC1	--	014:C fire suppr. bay 3 - open	--	5.1
RENDEZ RDR ANT PLB area	FLC1	FLC3	(c)	(c)	(c)
NLG backup uplk. release Nose whl. well area 21	FLC3	FLC2	LG ARM/ON reset sw. pnl. A12	PIC Logic	13.1
NLG strut thruster (ext) power assist Nose whl. well area 21	FLC3	FLC2	PIC logic	PIC logic	13.1
RMG backup uplk. release R whl. well area 62	FLC3	FLC2	PIC logic	PIC logic	13.1
LMG backup uplk. release L whl. well area 67	FLC3	FLC2	PIC logic	PIC logic	13.1
Ku-band ANT JTN PLB area	FLC1	FLC2	(c)	(c)	(c)
Docking tunnel PLB area	FLC1	FLC3	(c)	(c)	(c)

[a]Driver output located MEC No. 1.
[b]Driver output located MEC No. 2.
[c]To be flown TBD.

TABLE 7.6-III.- Continued

PIC	Location		How safed		SSSH dwg. no.
	Sys. A	Sys. B	Sys. A	Sys. B	
MEC Orb. PIC functions					
Orb./ET L aft sep.	MEC1	MEC2	MEC	MEC	5.2, 5.3
Orb./ET R umb. disc.					
Pyros 1A, 2A, 3A	MEC1		MEC	MEC	5.2, 5.3
Pyros 1B, 2B, 3B		MEC	MEC	MEC	5.2, 5.3
Orb/ET R aft sep.	MEC1	MEC2	MEC	MEC	5.2, 5.3
Mid. JTN cont. assy. PIC functions					
RH MANIP arm shldr. PLB area					
Pyros A (2)	Assy. no. 1		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 1		5A cb PNL ML86B:D	5.7
RH MANIP arm fwd. latch PLB area					
Pyros A (2)	Assy. no. 1		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 1		5A cb PNL ML86B:D	5.7
RH MANIP arm mid latch PLB area					
Pyros A (2)	Assy. no. 1		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 1		5A cb PNL ML86B:D	5.7
RH MANIP arm aft latch PLB area					
Pyros A (2)	Assy. no. 1		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 1		5A cb PNL ML86B:D	5.7

[a]Driver output located MEC No. 1.
[b]Driver output located MEC No. 2.

TABLE 7.6-III.- Continued

PIC	Location		How safed		SSSH dwg. no.
	Sys. A	Sys. B	Sys. A	Sys. B	
LH MANIP arm shldr.					
PLB area					
Pyros A (2)	Assy. no. 2		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 2		5A cb PNL ML86B:D	5.7
LH MANIP arm fwd. latch					
PLB area					
Pyros A (2)	Assy. no. 2		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 2		5A cb PNL ML86B:D	5.7
LH MANIP arm mid latch					
PLB area					
Pyros A (2)	Assy. no. 2		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 2		5A cb PNL ML86B:D	5.7
LH MANIP arm fwd. latch					
PLB area					
Pyros A (2)	Assy. no. 2		5A cb PNL ML86B:D		5.7
Pyros B (2)		Assy. no. 2		5A cb PNL ML86B:D	5.7
MED to GSE PIC functions					
R (LOX) T-0 umb. rel. Pyros A (1) Pyros B (1)	GSE pyro control rack 9A MLP	GSE pyro control rack 9A MLP			5.2
L (LH_2) T-0 umb. rel. Pyro A (1) Pyro B (1)	GSE pyro control rack 9A MLP	GSE pyro control rack 9A MLP			5.2

[a]Driver output located MEC No. 1.
[b]Driver output located MEC No. 2.

TABLE 7.6-III.- Concluded

PIC	Location		How safed		SSSH dwg. no.
	Sys. A	Sys. B	Sys. A	Sys. B	
R SRB holddown Pyros A (4) Pyros B (4)	GSE pyro control rack 9A MLP	GSE pyro control rack 9A MLP			
L SRB holddown Pyros A (4) Pyros B (4)	GSE pyro control rack 9A MLP	GSE pyro control rack 9A MLP			
L SRB sep. R SRB sep. L SRB ign. R SRB ign.	L SRB[a] (6) R SRB[a] (6) L SRB[a] (1) R SRB[a] (1)	L SRB[b] (6) R SRB[b] (6) L SRB[b] (1) R SRB[b] (1)	 Launch processor sys Launch processor sys		5.2, 5.3 5.2, 5.3 5.2, 5.3 5.2, 5.3
ET vent umb. ret.	ET vent arm panel	ET vent arm panel			
ET RSS	ET RSS distributor	ET RSS distributor	Manual safing and locking pin		5.5
Tumble sys.					5.3
Recovery sys. Nozzle ext. JTN	L SRB (1) R SRB (1)	--	None		5.4
Nose cap sep.	L SRB (1) R SRB (1)	--	None		5.4
Frustrum sep.	L SRB (1) R SRB (1)	--	None		5.4
Main parachute disc.	L SRB (1) R SRB (1)	--	None		5.4
RSS	RSS distributor (2)	RSS distributor (2)	Manual safing and locking pin		5.5

[a]Driver output located MEC No. 1.
[b]Driver output located MEC No. 2.

TABLE 7.6-IV.- THE PIC HISTORY TAB/DDD DISPLAYS

- Smoke detection/fire suppress.- 0633/0634 = History tabs

	Low	High	DDD format
V76V4736A BAY 1 PIC CAP V	-0.1	+0.2	30
V76V4716A BAY 2 PIC CAP V	-.1	+.2	30
V76V4700A BAY 3A PIC CAP V	-.1	+.2	30
- LG - 0619/0620 = History tabs			
V76V4820A EXT ACTR 1 NLG	-0.1	+0.2	30
V76V4821A EXT ACTR 2 NLG	-.1	+.2	30
V76V4830A EMER EXT A NLG	-.1	+.2	30
V76V4832A EMER EXT B NLG	-.1	+.2	30
V76V4900A EMER EXT A LMG	-.1	+.2	30
V76V4902A EMER EXT B LMG	-.1	+.2	30
V76V4950A EMER EXT A RMG	-.1	+.2	30
V76V4952A EMER EXT B RMG	-.1	+.2	30
- SRB/ET sep. seq. - 0663/0664 = History tabs			
V98X0744X BFS-SRB SEP ARM CMD	-	ARM	60
V98X0752X BFS-ET SEP ARM CMD	-	ARM	60
- RSS - 0661/0662 = History tabs			
B55V1623C LH PIC CAP A V	35.7	40.75	-
B55V1624C LH PIC CAP B V	35.7	40.75	-
B55V2623C RH PIC CAP A V	35.7	40.75	-
B55V2624C RH PIC CAP B V	35.7	40.75	-
T55V1730A ET PIC CAP A V	35.7	40.75	-
T55V1731A ET PIC CAP B V	35.7	40.75	-
B55X1877X LH RSS A ARM	-	ARM	60
B55X1878X LH RSS B ARM	-	ARM	60
B55X2877X RH RSS A ARM	-	ARM	60
B55X2878X RH RSS B ARM	-	ARM	60
T55X1931E ET RSS ARM	-	ARM	60
B55X1879X LH RSS A FIRE	-	FIRE	-
B55X1880X LH RSS B FIRE	-	FIRE	-
B55X2879X RH RSS A FIRE	-	FIRE	-
B55X2880X RH RSS B FIRE	-	FIRE	-
T55X1933E ET RSS FIRE	-	FIRE	-
B55X1870X LH RSS S&A ARM	-	ARM	-
B55X2870X RH RSS S&A ARM	-	ARM	-
T55X1870X ET RSS S&A ARM	-	ARM	-

7.7 ADVANCED SOLID ROCKET BOOSTER

7.7.1 ASRB Overview

The purpose of this brief is to provide a concise summary of the advanced solid rocket booster (ASRB) hardware. In particular, this summary focuses on the major differences between the RSRB and the ASRB. At the time of this writing, the future of the ASRB program is in doubt as a result of funding constraints at NASA. The total program research, development, and testing cost is about $2.5 billion (1991 dollars). If full funding is approved and the development schedule remains constant, the first flight of the ASRB will occur in February 1997. As a result of the development flight instrumentation (DFI) integration constraints, the first ASRB flight will most likely be assigned to OV-102 (Columbia).

The ASRB program is being managed by MSFC with Lockheed/Sunnyvale as the principal integrator. Major subcontractors include Aerojet, Rust Engineering, USBI, Babcock & Wilcox, Ladish, and Thiokol. The ASRB will be assembled at the government-owned, contractor-operated (GOCO) Yellow Creek site in Iuka, Mississippi. Full-scale, full duration test firings will be at Stennis Space Center near Bay St. Louis, Mississippi. Nozzles will be assembled by Thiokol at Michoud, Louisiana. Aerojet, of Sacramento, will provide the design of the case, igniter, initiator, and propellant grain structure. Case forgings will arrive from Ladish in Wisconsin. USBI will integrate and refurbish the ASRB electronics, aft skirt, and thrust vector control/hydraulic system. Either WECCO or Kerr-McGee, both of Nevada, will provide the ammonium perchlorate for the propellant oxidizer. It should be noted all design items mentioned here are preliminary, as this document was written prior to the ASRB CDR.

7.7.2 ASRB Purpose

The purpose of the ASRB is twofold, to provide a greater lift capacity for shuttle to orbit and to improve the design to provide additional safety and reliability.

The ASRB is designed to provide for about 12,000 lb of extra performance to low-Earth orbit per launch (6000 per ASRB). Not all of this performance can be translated into extra payload capacity because the orbiter cannot land with more than about 55,000 lb of payload without a waiver or design changes to meet a greater landing weight capability. Much of this extra performance will be used to widen abort boundary windows and to reduce abort gaps for dual engines out. In addition, the thrust versus time slope will be augmented from the RSRM to avoid the need to throttle the SSME's in first stage below 100 percent power level. This will reduce the hazards associated with the main engines stuck at a low power level in first stage from an electrical or hydraulic lockup.

7.7.3 ASRB Design Changes

7.7.3.1 Case

The ASRB is the same length as the RSRB, 1513 in. (126 ft), but has a wider diameter, 150 in. rather than 146 in. This increase of 4 in. allows extra propellant volume for higher first-stage impulse. The liftoff weight will increase from 1,256,000 lb in the present design to 1,348,000 lb. The steel case will be in three equal length segments (fig. 7.7-1) rather than the four segments with the RSRM. The case will be 9Ni-4CO-0.3C medium-alloy steel rather than D6AC steel as with the RSRB. This new alloy will have higher strength, be more weldable, and offer greater corrosion resistance than D6AC. Nine stiffener rings in the aft segment will be integral with the metal case wall rather than bolt-on additions. The ET attach ring will also be integral with the aft segment case wall.

7.7.3.2 Case Joints

The field joint design will be quite distinct from the RSRB. An axial bolted design will be used to attach the three segments together with two field joints (fig. 7.7-2). A low temperature qualified Viton, Viton GLT, will be used to form the three polymer seals at each field joint. This will eliminate the need for field joint heaters necessary for the Viton A O-rings in the present RSRB. The layer of case wall insulation adjacent to the field joint will use a bonded pressure activated J-seal similar to the present design. The 11 case factory joints will be welded together rather than incorporate a pin-clevis-tang design. The welds will be of the straight polarity plasma arc type.

7.7.3.3 Propellant

The ASRB will use a propellant that is slightly more energetic and more dense than the RSRB PBAN mix. This propellant is known as hydroxy-terminated polybutadiene (HTPB). The propellant will be a mix of 88 percent solids (i.e., ammonium perchlorate and a polymer binder) together with 19 percent spherical aluminum and a ferric oxide (rust) burn rate catalyst. The new propellant will offer better processability and higher structural margins than the RSRM propellant. The motor mass fraction at liftoff (W prop/W total) will rise from 0.882 in the RSRB to 0.894 in the ASRB. Vacuum specific impulse will go from 267.9 to 270.3 seconds. A soluble core process will be used to form the forward propellant star pattern shape. A continuous mix process with constant sampling will be used for propellant production. The Thiokol RSRM facility uses a batch method for propellant production.

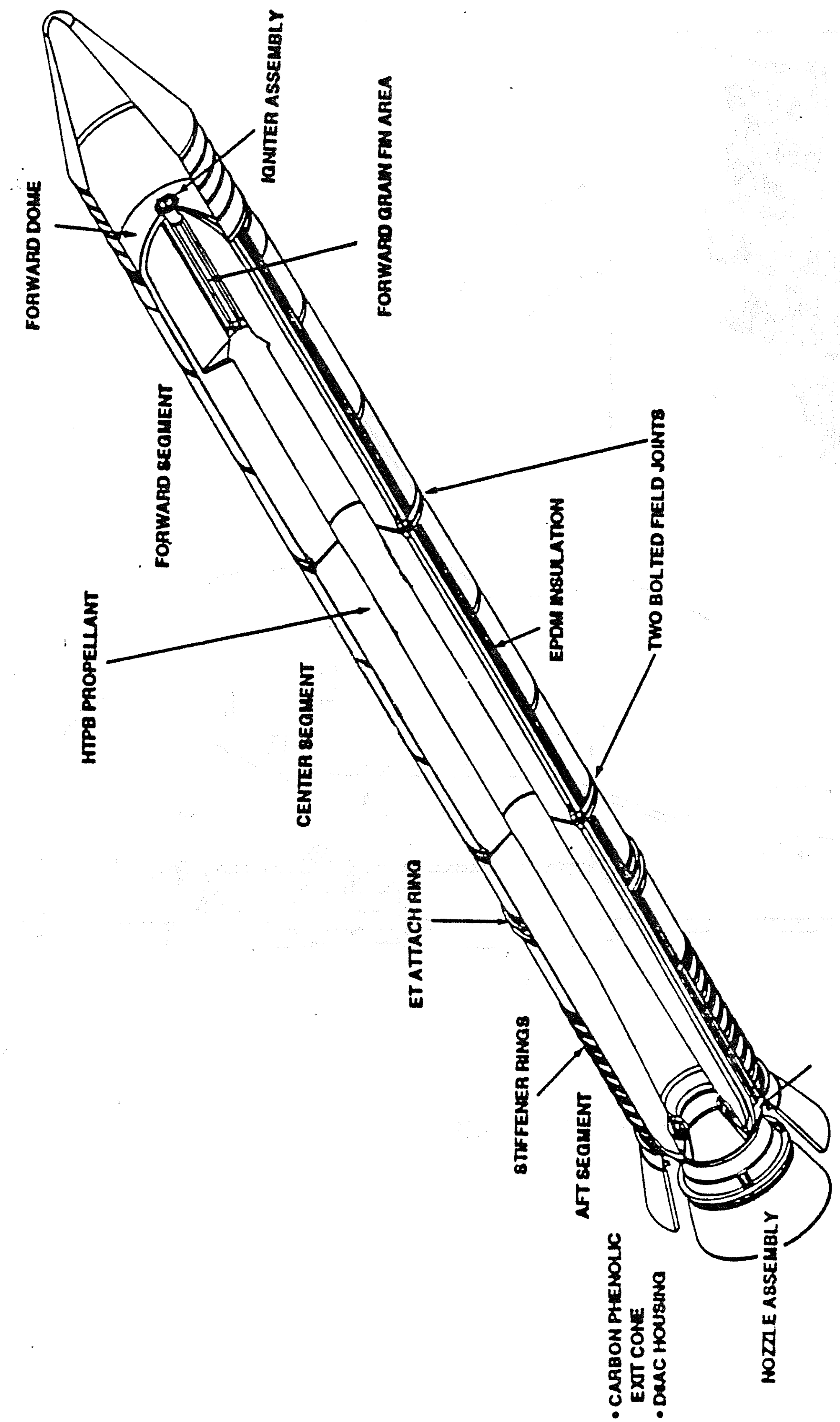

Figure 7.7-1.- ASRB configuration.

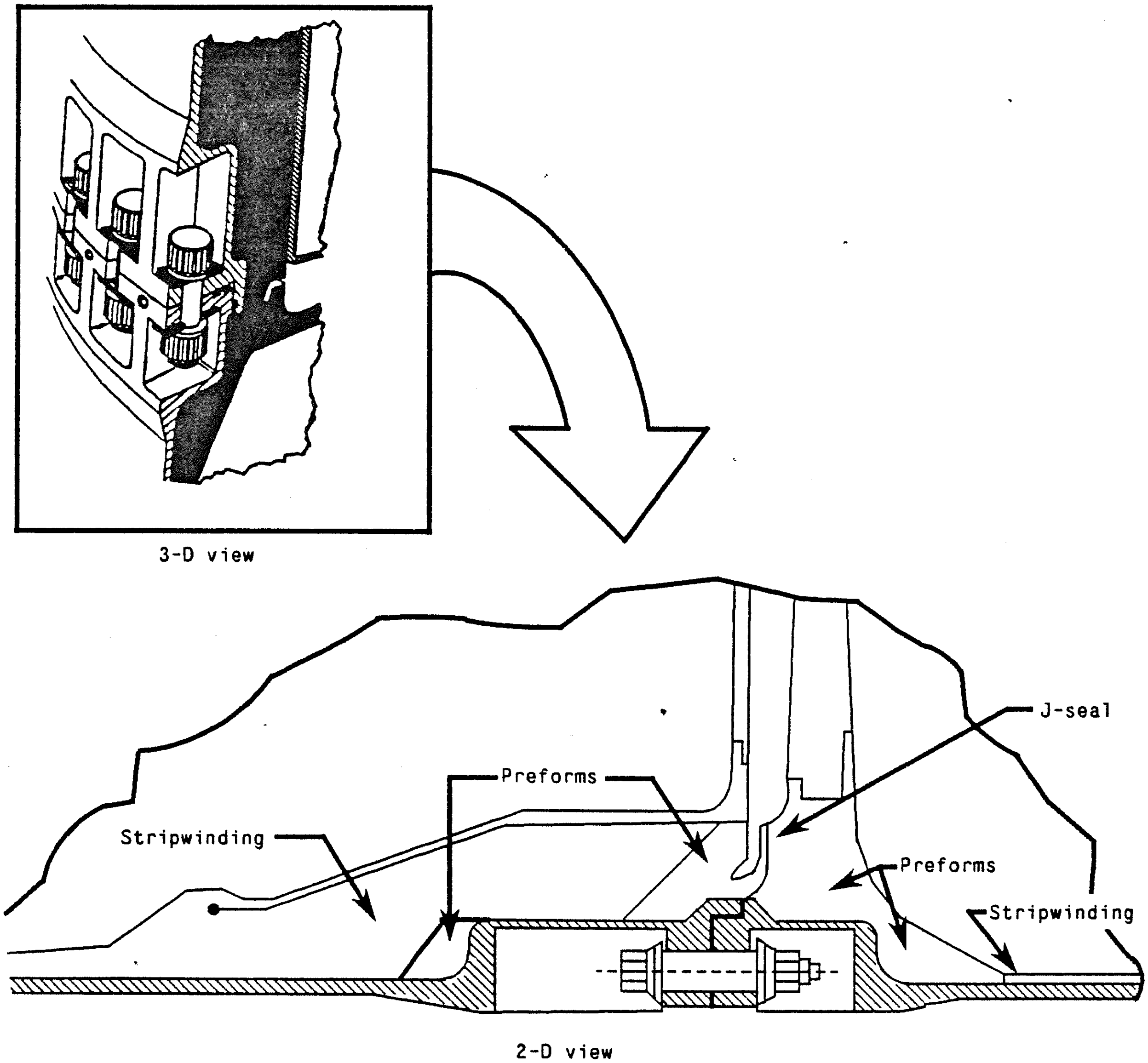

Figure 7.7-2.- ASRB field joint.

7.7.3.4 Case Insulation

The case wall insulation will use a Kevlar-glass EPDM polymer matrix. The present RSRM uses an asbestos-silica filled NBR. The EPDM will be totally asbestos free.

7.7.3.5 Nozzle and Nozzle-to-Case Joint

The ASRB nozzle will incorporate a carbon-carbon phenolic (CCP) inner liner for greater strength and heat resistance. The new design will use fewer parts (fig. 7.3-3) as well. The nozzle will be movable about a flexible bearing of a different design than the present bearing, although the function will be the same. The thrust vector control (TVC) system should be similar. The number of critically 1R leak paths in the nozzle will reduce from 82 to 9. The simpler new design will eliminate the flex seal and cowl and boot assembly. This will also delete the vented cowl/boot cavity. Two of the five RSRM internal joints will be eliminated. Support structure for the carbon phenolic ablatives will be a mix of 7050 aluminum and D6AC steel. The nozzle ablatives will be of two separate types, standard density and low density CCP.

The nozzle-to-case joint will be a new design also. It is presented on figure 7.7-4.

7.7.3.6 Instrumentation

A new design will be used for the ASRB pressure transducers. As with the RSRB, three transducers will be mounted in the forward dome region adjacent to the igniter. The new design will allow for greater future parts availability and should be of equal or better reliability than the present transducers. No changes are expected to any console displays, MSID's, or telemetry needed by the booster, MPS, or ME console operators from the switch from RSRB to ASRB.

Extensive DFI will be used on the first few flights of the ASRB to measure aero loads, thermal effects, structural loads, and other performance parameters.

7.7.3.7 Igniter

As with the RSRB, the ASRB will incorporate an initiator and a similar sized igniter. The igniter will be multiport design rather than a single-axial port as with the RSRB. The propellant will be a cartridge-loaded HTPB mix. See figure 7.7-5.

ASRM NOZZLE DESIGN

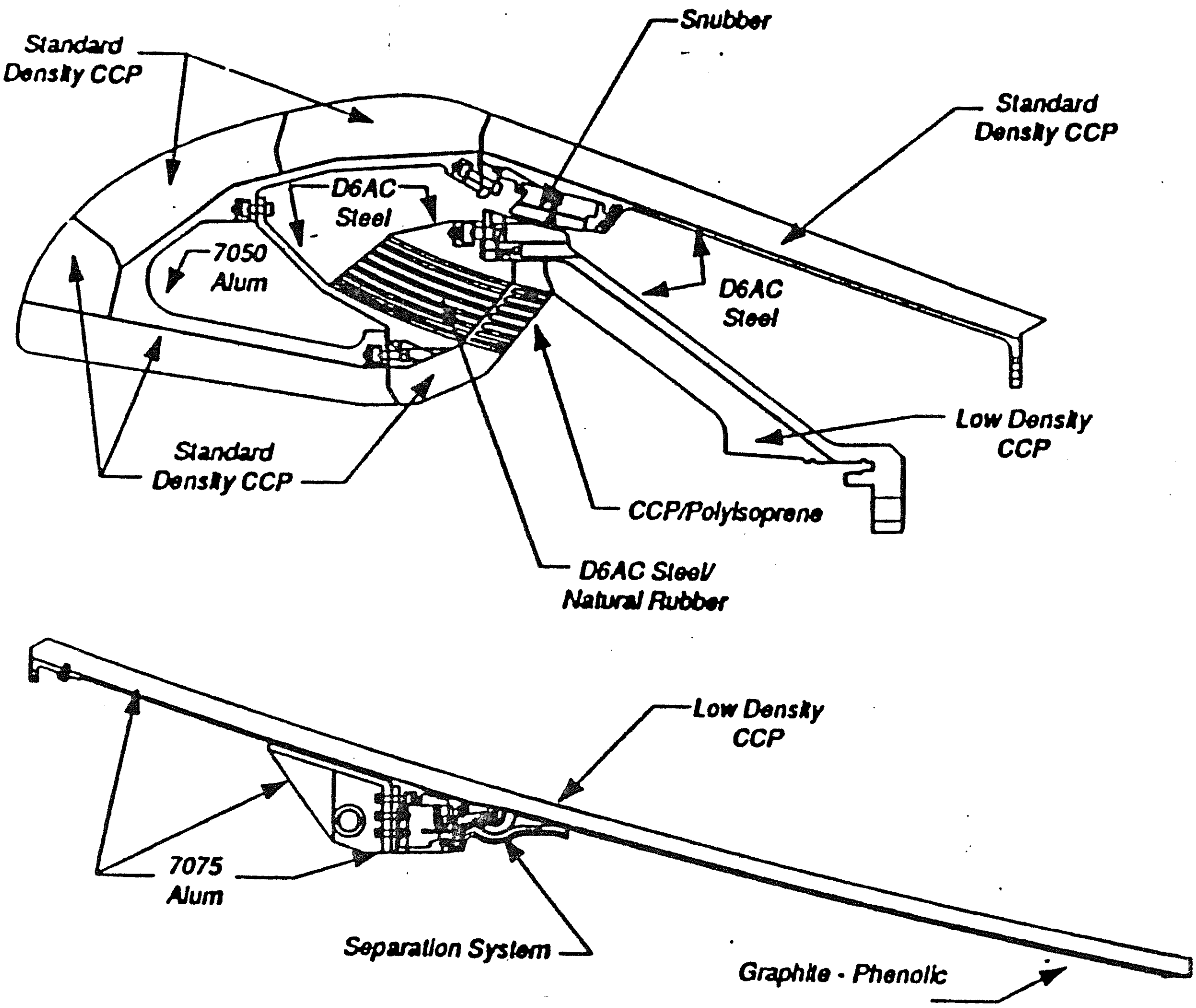

Figure 7.7-3.- ASRB nozzle.

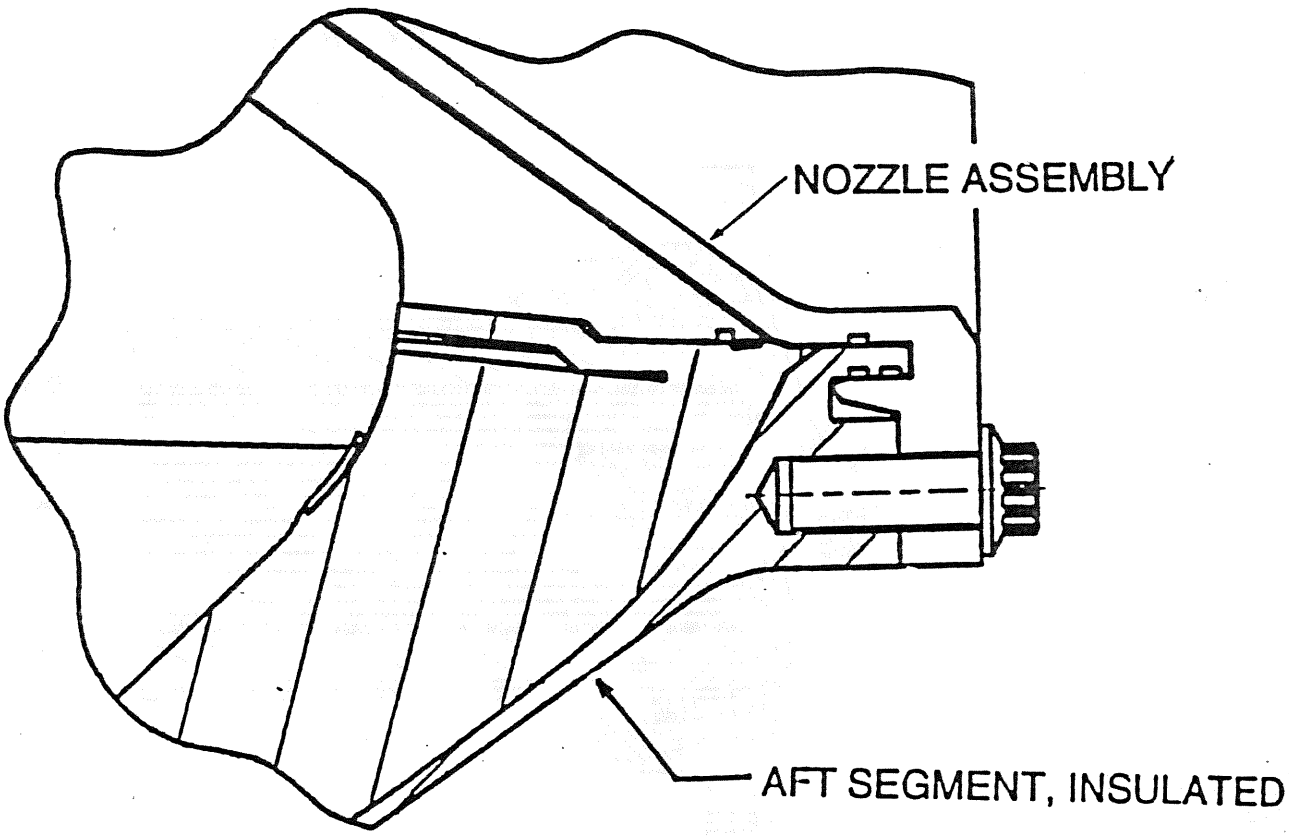

Figure 7.7-4.- Nozzle-to-case joint.

Figure 7.7-4.- Nozzle-to-case joint.

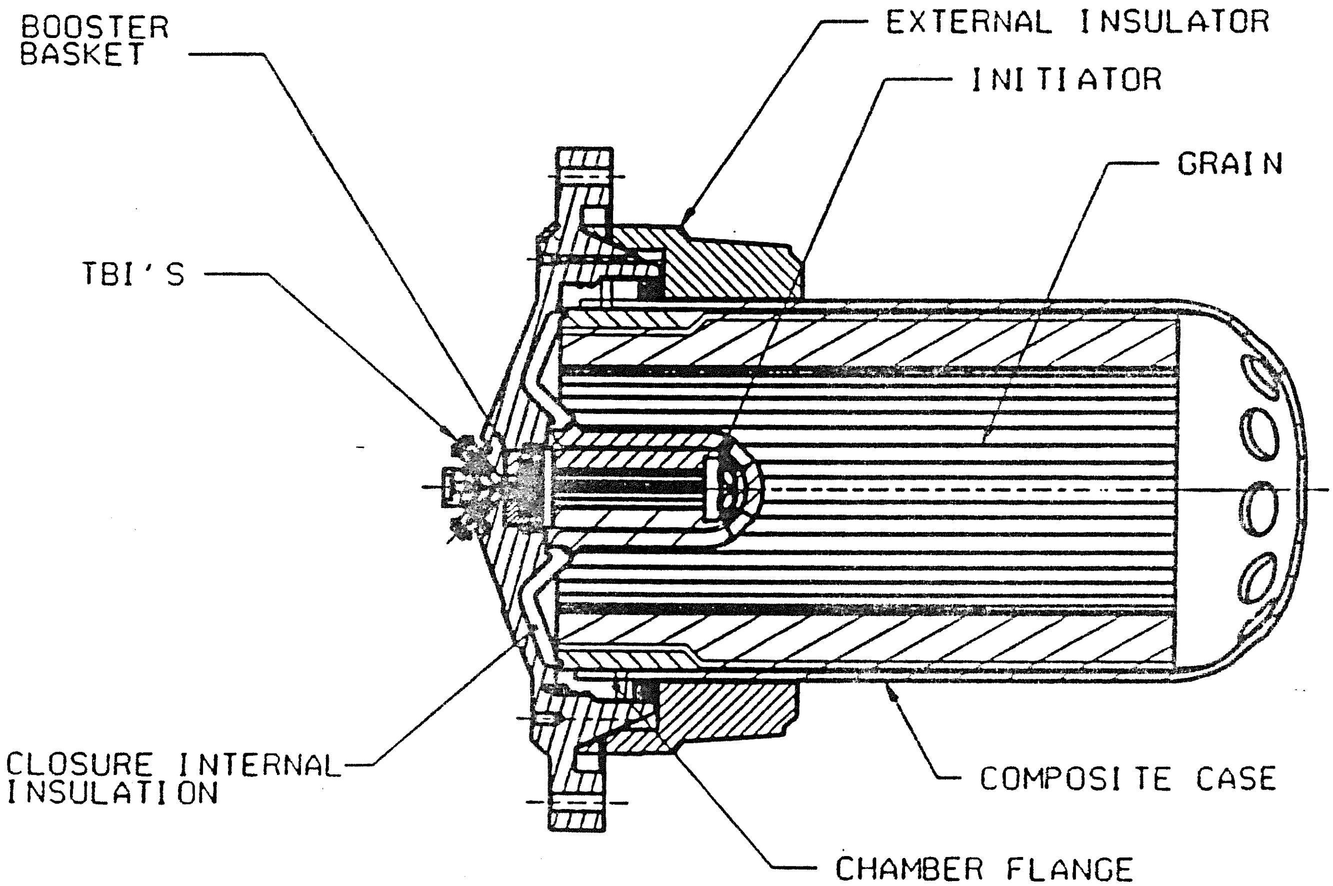

Figure 7.7-5.- ASRB igniter.

7.7.3.8 Electronics/Avionics

The electrical and avionics system of the ASRB will be similar to the RSRB. The component level parts may be of a different construction and design but their function and interconnections should be very similar. The aft integrated electronics assembly (IEA) may be relocated to a different position to alter ascent drag and bending moment loads applied to the ASRB.

7.7.4 ASRB Performance and Flight Operations

Compared to the RSRB, the ASRB will have a higher initial thrust, lower thrust during the maximum dynamics pressure timeframe, and higher thrust for a longer duration at the tail end of the propellant burn (fig. 7.7-6). Nominal ASRB separation will occur later, at about 2:16 (min:sec) MET, at a higher altitude of about 180,000 feet. Nominally, the RSRB separates at 2:06 at 150,000 feet.

The higher performance levels will enable the orbiter to have overlapping RTLS and ATO boundaries, negating the need for a single engine out TAL case, if desired. Negative return is expected to be between 3:35 and 3:45 MET for most missions. Conversely, the earliest single engine out TAL case will occur much sooner, probably about 2:16 MET (ASRB sep time), decreasing the RTLS window of vulnerability, if desired. Without the SSME throttle bucket, the performance penalties associated with a hydraulic or electrical lockup early in flight will be minimized.

The penalty for the extra performance includes higher aerodynamic and thermal loads during first stage, and much higher reentry loads after ASRB separation. Extra thermal insulation protection may be required for the aft end of the orbiter and ET as a result of increased plume heat radiation. Analysis is in work for a recalculation of structural loads and safety factors for the orbiter and ET in first stage.

No major post-MECO or ARD impacts are expected to occur with the ASRB flights that result from RTLS/TAL/AOA aborts.

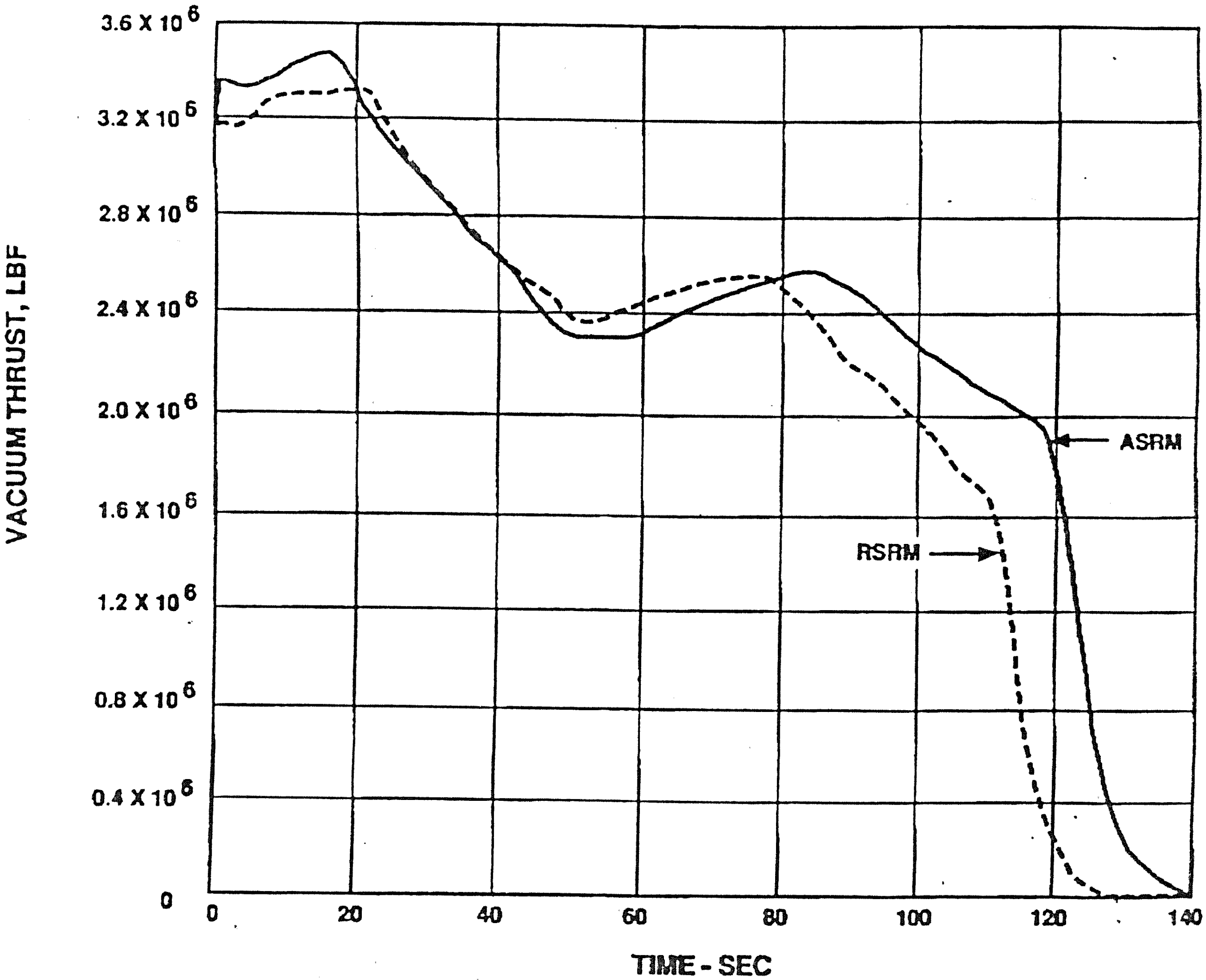

Figure 7.7-6.- RSRM/ASRM thrust-time plot.

APPENDIX A

BOOSTER SYSTEMS BRIEFS
ADVANCED/HISTORICAL PROJECTS
OCTOBER 1, 1992

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1.7 Two Duct Hot Gas Manifold

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SECTION 1: SSME

1.1 Alternate Turbopumps
1.2 SSME 10K Pumps
1.3 Block II Controller
1.4 Wide Throat Main Combustion Chamber
1.5 Single Tube Heat Exchanger
1.6 Phase II+ Powerhead
1.7 Two Duct Hot Gas Manifold

1.1 Alternate Turbopumps (Pratt and Whitney)

Function

The high pressure fuel turbopump (HPFT) and high pressure oxygen turbopump (HPOT) are identified to be upgraded by the alternate turbopump development (ATD) program. The HPFT boosts the fuel pressure from the low pressure fuel turbopump to the engine operating level and supplies hydrogen to the engine fuel and cooling circuits. The HPOT consists of two pumps on a common shaft. The main pump boosts the LO_2 pressure from the low pressure oxygen pump and supplies oxidizer to the main combustion chamber, the heat exchanger, the LPOT turbine, and the preburner oxidizer pump. The preburner oxidizer pump boosts the pressure to a higher value which supports the injection of LO_2 into the fuel and oxidizer preburners.

Purpose

The (ATD) design approach utilizes new technology available since SSME turbopumps were designed. The intent of developing both the HPFT and HPOT ATD's is to minimize operational cost and increase performance margin through innovation in design, producibility, inspectibility, and maintainability. The goal is to achieve 55 mission/7.5 hours of life. In addition the new design will use conservative, inspectable designs and design technology such as fracture mechanics control, finite element structural analysis, materials characterization, rotordynamics analysis, computational fluid dynamics, rig and model verification at the component and part level. The design and development was conducted by Pratt-Whitney, the integration support by Rocketdyne, the component testing at Pratt-Whitney, and the certification and flight certification extension testing at Stennis Space Center.

The ATD's will address several of the CIL's with the following identified corrective actions. Structural failure of the turbine blades will be addressed using cored airfoils, and reduced blade stresses reduced using a 2 lobe firtree. The loss of support or position control will be resolved by employing a simplex bearing, roller turbine bearing, and a single piece stiff rotor. Loss of coolant flow to the fuel turbine will be alleviated by relocating the turbine bearing to the inboard position in which coolant is always provided to the bearing. The loss of coolant flow to turbine disk is resolved with a positive coolant feed system to the turbine disk.

Results of design tests and extensive post test data and hardware analysis provide the basis for improving the design and prediction models. Tests have achieved 111% steady-state power conditions on the fuel turbopump and 104% steady state and 100%

transient power conditions on LOX turbopump with liquid nitrogen.

Alternate High Pressure Oxidizer Turbopump

The HPOT design utilizes single crystal turbine blades. A double collecting volute will reduce bearing side loads. Welds are minimized using fine grain castings. The rotor system is stiffened, the rotating elements are reduced to 28 total, and the bearings are minimized i.e. 2 ball bearings and 1 roller bearing. The new design provides significant suction margin i.e. 40%. The pump-induced radial load is reduced to 500 lb_f. There will be only one LOX cooled bearing, and turbine sheet metal is eliminated by using thinwall, thermally compliant, fine grain castings instead. Figure 1 shows the new HPOT pump compared to the current SSME, and Figure 2 shows the cutaway of the ATD HPOT. The HPOT ATD will improve reliability, producibility, reduce fabrication and operational costs, improve mean time between failure, and improve the turnaround time at KSC.

HPOT Current Status

The ATD LOX pump 05-1 was tested at E8 to a maximum power level of 104%. No significant anomolies have been identified. The pump is presently at Stennis Space Center (SSC) to be installed on stand B-1 and tested with the HPFT F54.

Alternate HPFT

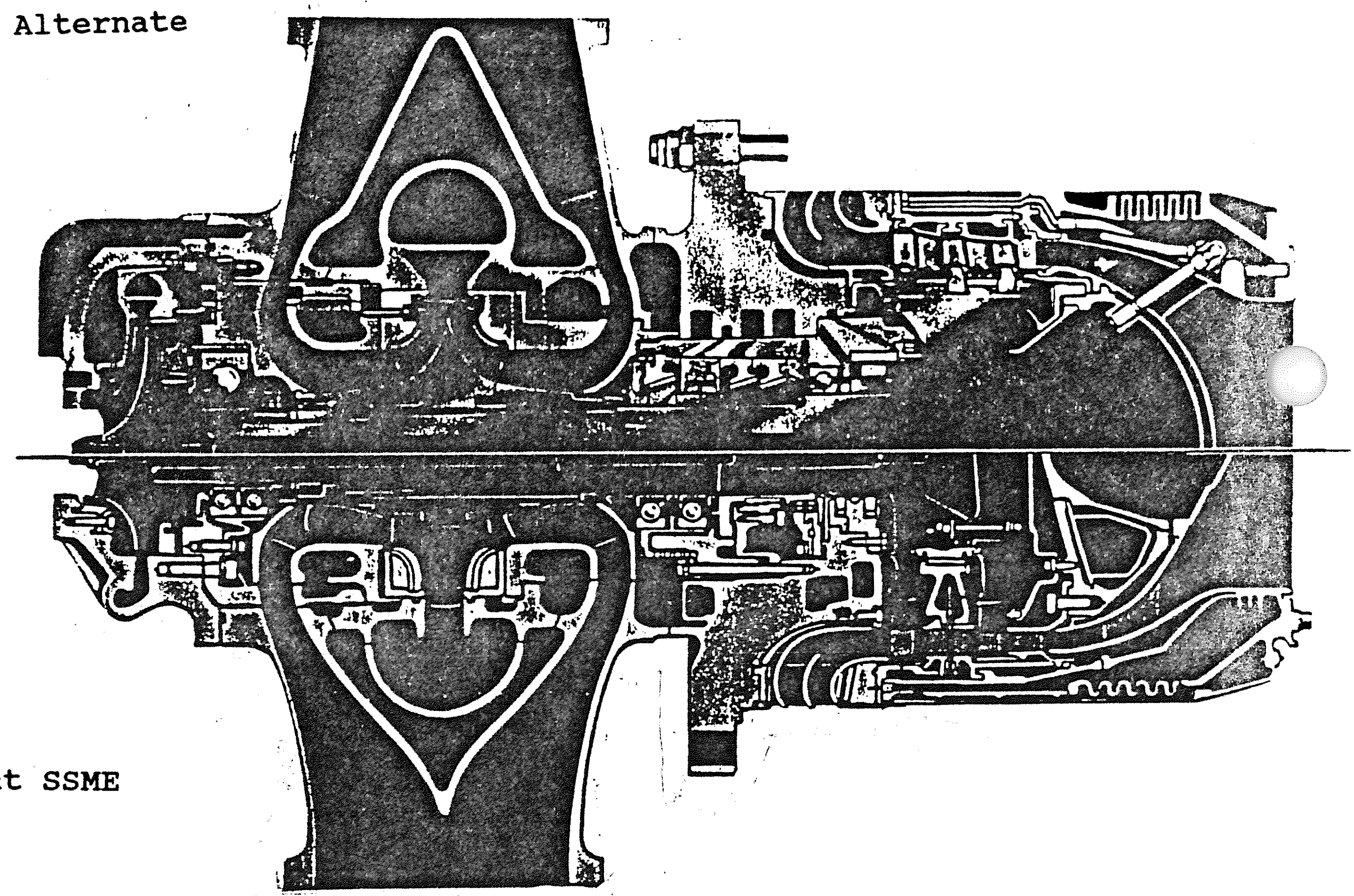
The HPFT design emphasizes durability and operability by incorporating pump hydrodynamic margin and stability in the design. A double acting rotor is used for thrust balance control. Proven turbine materials i.e. single-crystal turbine blades are used which eliminate the thermal barrier coating. Welds are eliminated using fine grain precision castings. The rotor system is stiffened, rotating elements are reduced to 14, and there is only two bearings i.e. 1 ball bearing and 1 roller bearing. 90% suction margin is provided for. Turbine sheet metal is eliminated using thinwall fine grain castings. The inlet housing strength is increased for surge pressure loading. Figures 3 and 4 show the ATD compared with the HPFT SSME and the ATD HPFT cutaway respectively.

HPFT Current Status

The ATD fuel pump F5-1 has tested at SSC for 5 tests and 880 seconds. A maximum power level of 104% has been achieved. At present no significant anomolies have been identified.

HPOTP COMPARISON

P&W Alternate



Current SSME

Figure 1

HOWMET PRECISION CASTINGS IN PRATT & WHITNEY ATD HPOTP

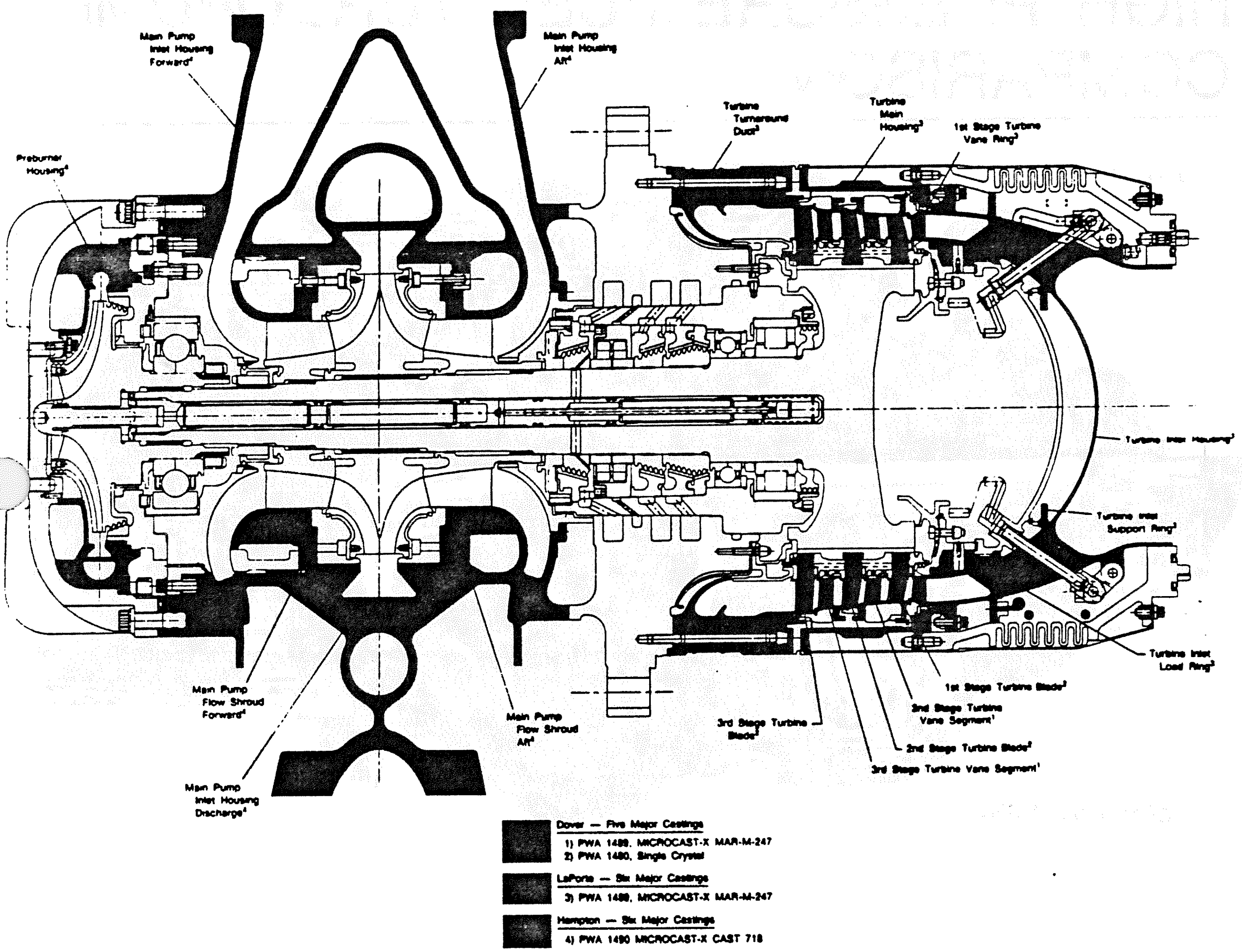
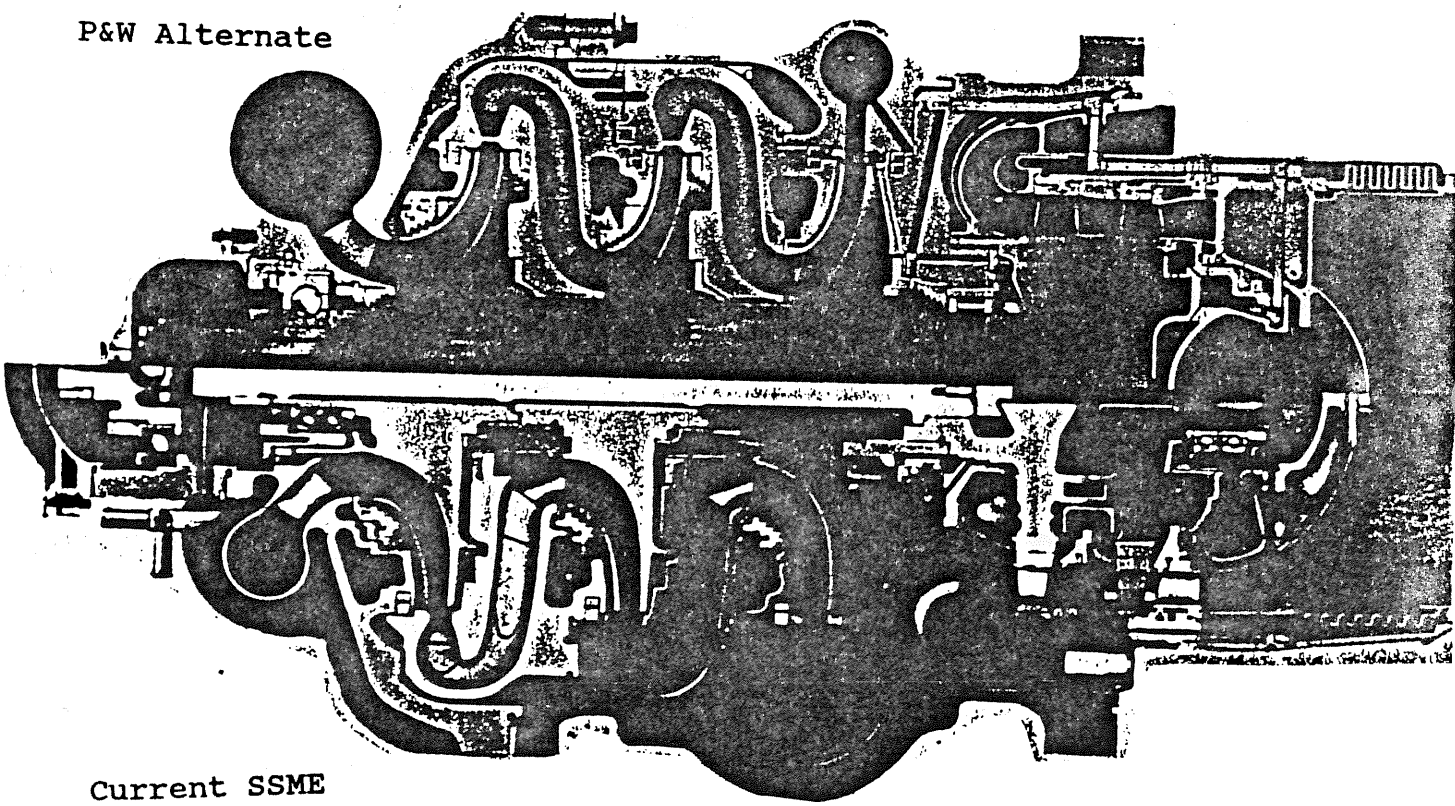


Figure 2

HIGH PRESSURE FUEL TURBOPUMP/ COMPARISON

P&W Alternate



Current SSME

Figure 3

HOWMET PRECISION CASTINGS IN PRATT & WHITNEY ATD HPFTP

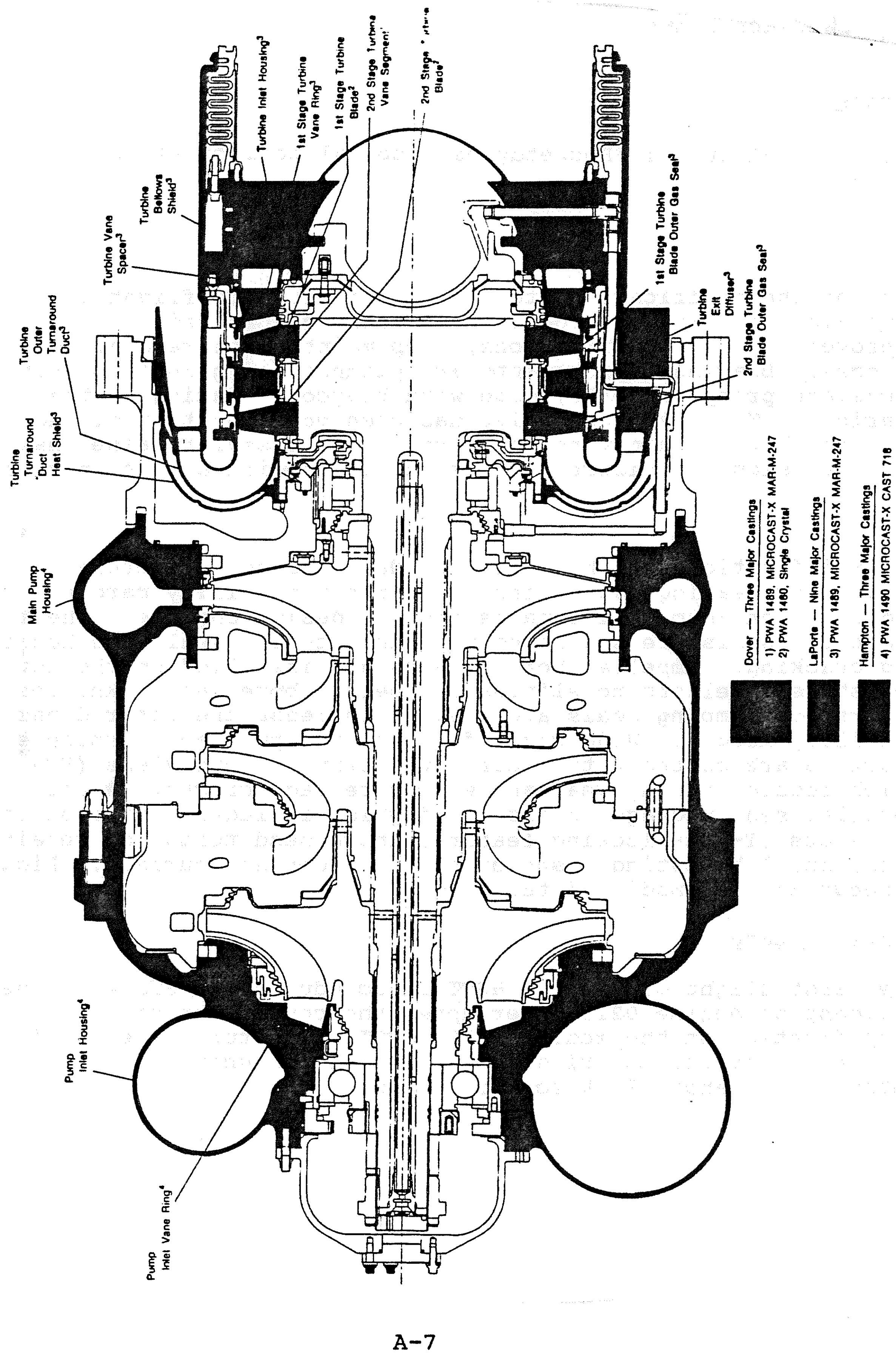


Figure 4

1.2 SSME 10K Pumps

Purpose

The SSME 10K pump is Rocketdynes proposal to improve the HPFT and the HPOT.

HPOT

Some of the modifications to the HPOT include in-flight and postflight bearing wear detection, bearing load reduction, improved cooling modifications, pump maintenance removal design changes. Bearing improvements are planned using an fluorinated ethyelene propylene (FEP) cage with braycote coating on the ball bearings. The five vane inlet has been modified to a fifteen vane inlet with thin vane inducers. The tip seal retainer has been redesigned. Figure 5 shows the 10K modifications in detail.

HPFT

The modifications to the HPFT include increased structural capability bearing cages, increased rotor stability margin modification, pump maintenance removal design changes. The first stage tip seals are scalloped to eliminate tip seal and housing lug cracking. Impeller bore inserts are installed on the 1st and 2nd stage impellers to eliminate impeller bore wear. Knurled interstage damping seals are used to increase the rotor dynamic stability margin. Wide cages for both the pump and turbine end bearings are coated with fluorinated ethylene propylene (FEP) which reduce bearing wear and eliminate cage cracks. A Kel-F impeller seal package is installed which includes a widened lock tab slots, T-lock locking feature, increasesd torque on retaining nuts, and lubricating paste on the shim contact surface. Figure 6 shows the 10K modified turbopump.

Current Status

The first flight of the 10K HPFT is scheduled for STS-45. The incident on engine 0215 interrupted the completion of certification on the modified 10K HPOT. The turbine end suffered damage due to the LOX rich shut down and the unit has been returned to Canoga Park for disassembly.

HPOTP 10K CONFIGURATION

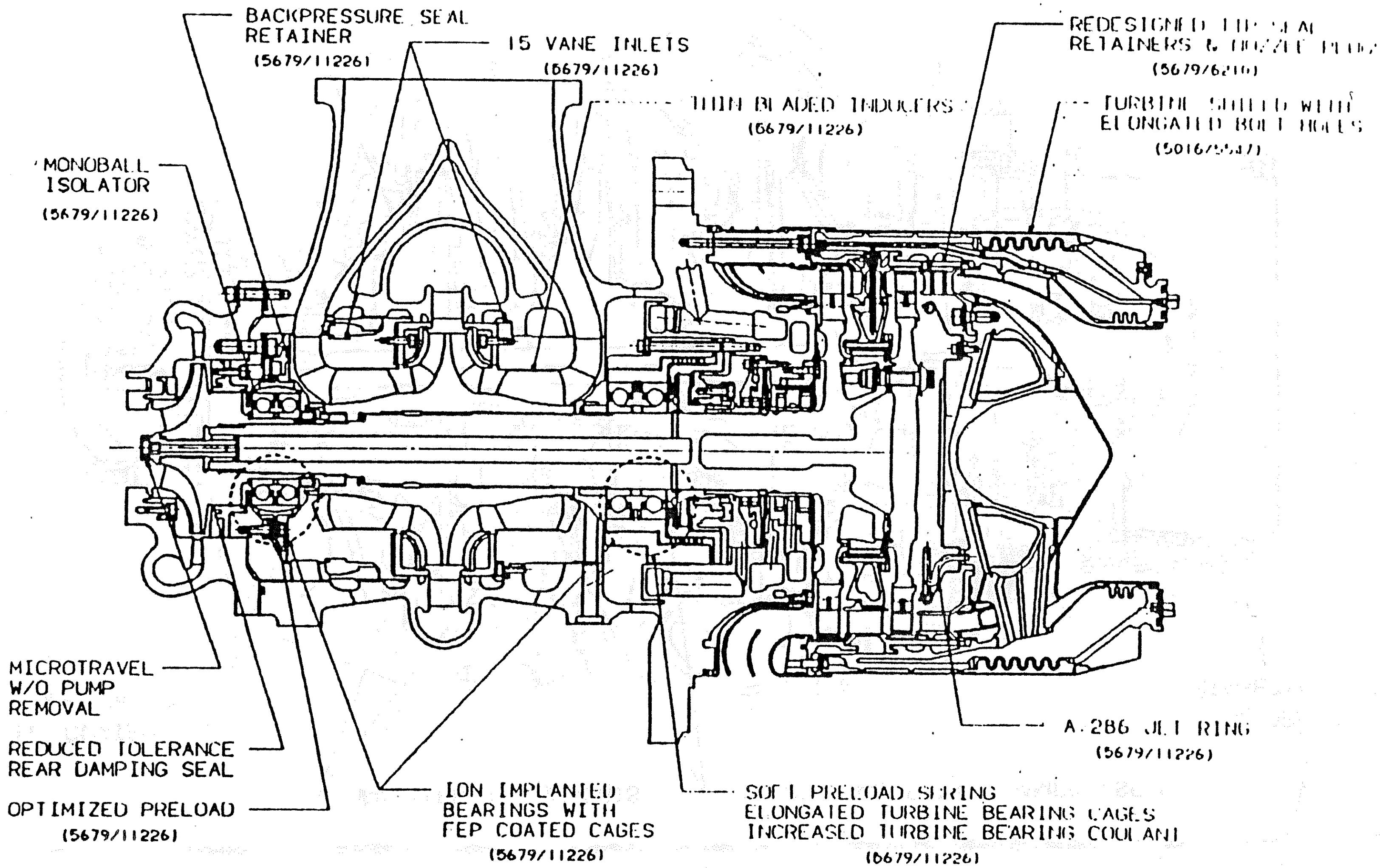


Figure 5

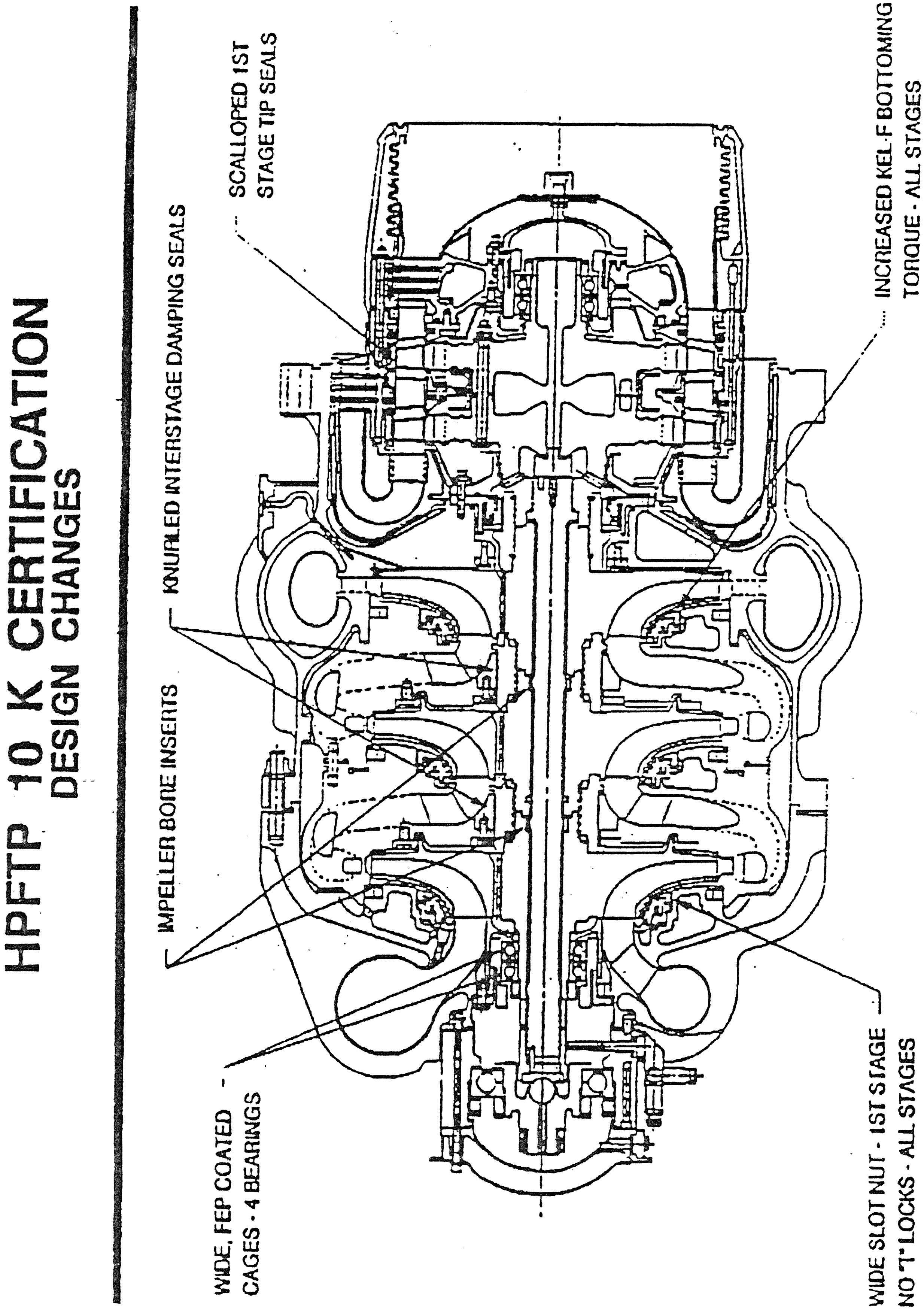

HPFTP 10 K CERTIFICATION
DESIGN CHANGES
SCALLOPED 1ST STAGE TIP SEALS
KNURLED INTERSTAGE DAMPING SEALS
IMPELLER BORE INSERTS
WIDE, FEP COATED CAGES - 4 BEARINGS
INCREASED KEL-F BOTTOMING TORQUE - ALL STAGES
WIDE SLOT NUT - 1ST STAGE
NO "L" LOCKS - ALL STAGES

Figure 6

1.3 Block II Controller

Function

The SSME controller is a single, integral electronics package mounted on the SSME. The controller is specifically designed to operate in conjunction with engine sensors, valves, actuators, spark ignitors, harnesses, and the operational computer program which does engine checkout, control, and monitoring. The controller is packaged in a sealed, pressurized chassis with cooling provided by convection heat transfer through pin fins.

Comparison

The SSME Block II Controller is the second generation Main Engine Controller replacing the current Block I controller. Upgrading to the Block II is necessary due to the obsolescence of electrical components that are based on early 1970's technology. The hardware comparisons are as follows:

The Block II functions with the SSME in an identical fashion as the Block I Controller. The size is identical to the Block I Controller and the weight is less than or equal to the Block I.

The memory on the SSMEC is a solid plated wire and is fixed memory at 16K. The Block II controller has 64K RAM + 2K PROM and therefore is more susceptible to disruption.

The FASCOS on the SSMEC are external and have been disconnected by disconnecting the hard wires. The FASCOS were being used for engine shutdown due to high vibrations but are now used for information only. The FASCOS will be internal to the Block II Controller and will be a software command instead of hard wired.

The Block II Controller interface with the Orbiter requires wiring changes in the 28 VDC power circuit, on the engine side of the interface. The 28 VDC is currently routed to the existing FASCOS. In the Block II controller, this 28 VDC is needed to provide backup electrical power support for the volatile memory during flight. The GPC software which currently turns the 28 VDC off during the MPS dump sequence after MECO, requires revision to maintain the 28 VDC throughout the flight.

Hardware Comparison Between SSMEC and BLOCK II

SSMEC	**Block II**
H 602 CPU	MC68000 microprocessor self checking pairs

1.5% temp sensor accuracy	1.0% temp sensor accuracy
SSMEC	**Block II**
16K words memory capacity plated wire	64K words memory capacity 2K PROM
direct memory access	dual port memories
600 watts power consumption	400 watts power consump.
hot gas temp sensors fail into qualified zone	hot gas temp sensors fail into disqualified zone
external fascos box	fascos internal to controller
computer internal electronics B can only supply excitation to digital electronics B	computer internal electronics B can supply excitation to both A&B digital electronics
240 lbs	210 lbs
91 sensor/monitor inputs	106 sensor/monitor inputs
$7.0M	$2.5M

1.4 Wide Throat Main Combustion Chamber (MCC)

Function

The main combustion chamber contains the combustion process, accelerates the gas flow to throat sonic velocity, and initiates the gas expansion process through its diverging section. The chamber includes a liner, jacket, throat ring, coolant inlet manifold, and coolant outlet manifold.

Comparisons

The larger throated MCC has several specific objectives inherent in the design. The new design uses fine grained investment casting and vacuum plasma spray/liquid interface diffusion bonding with eliminates 56 of 58 critical welds,and the remaining welds are inspectable. There is a 10% reduction in system operating pressures which allows increased margins. An integral coolant liner/structural jacked eliminates a CRIT 1 liner implosion failure mode. Hydrogen embrittlement sensitive materials are eliminated. The coolant liner wall temperature is reduced thereby extending the MCC life. Less maintenance and inspections will be required, and the speeds and torques are reduced. There is a possiblity of a specific impulse penalty, however the preliminary test shows little or no penalty. There could also be susceptability to combustion instability. An extensive bomb test program was conducted with no instability evident. The wide throat MCC requires a low pressure oxygen turbopump (LPOT) inducer modifications. This was confirmed by a hot fire. A planned thin vane inducer/15 vane inlet modification to the LPOT eliminates the concern.

1.5 Single Tube Heat Exchanger

Function

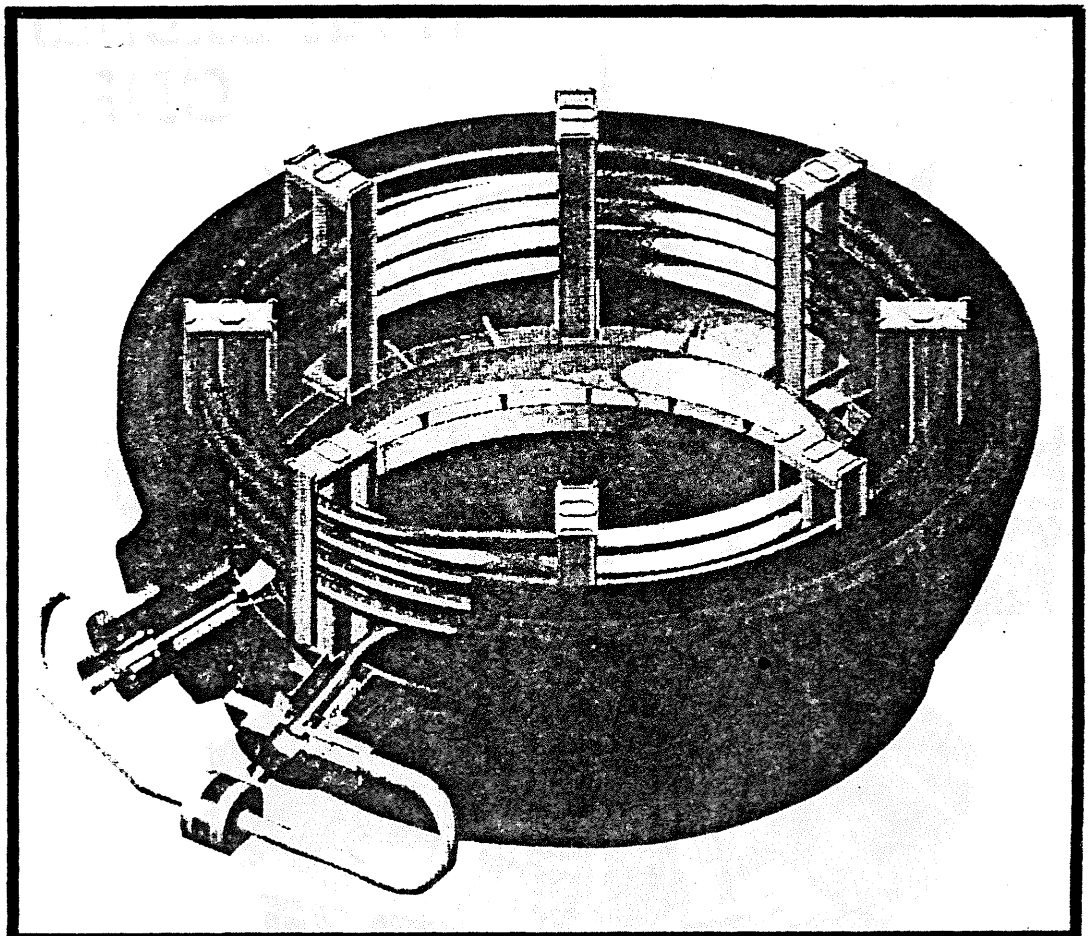
The oxygen heat exchanger supplies gaseous oxygen to pressurize the oxygen tank to maintain gaseous ullage pressure. Nominal performance is 1.55 lb/sec and 850 ^{o}R. The heat exchanger also supplies pogo pressurization to prevent propellant surge.

Comparisons

The new design incorporates features to eliminate problem areas in the current bifurcated HEX coil design. These include coil weld failures and tube wear damage. The new design eliminates seven interpropellant coil welds, and the thick tube wall increases margin for tube wear and damage. The tube wall is increased from 0.0125 and 0.0265 to a uniform 0.032. Welding is reduced at the inlet and outlet port areas and the assembly sequence is simplified. Thermal expansion loops are incorporated on the inlet end of the tube. The overall tube length is increased from 30 to 41 feet. Figures 7 and 8 show what the HEX will look like.

Current Status 11 Jul 91

The first production unit has been successfully coiled into its final configuration. During bonding of the inlet portion of the coiled tube a .040 deep wrinkle was created in the tube wall. Structural analysis is in work to determine if wrinkle rework is necessary. The single tube heat exchanger will increase GO_2 supply pressure by 250 psi. This will require resizing the GO_2 fixed orifice. Preliminary assessment indicates 74% to maintain requirements. A decision is pending actual test data.



Single Tube Heat Exchanger

Figure 7

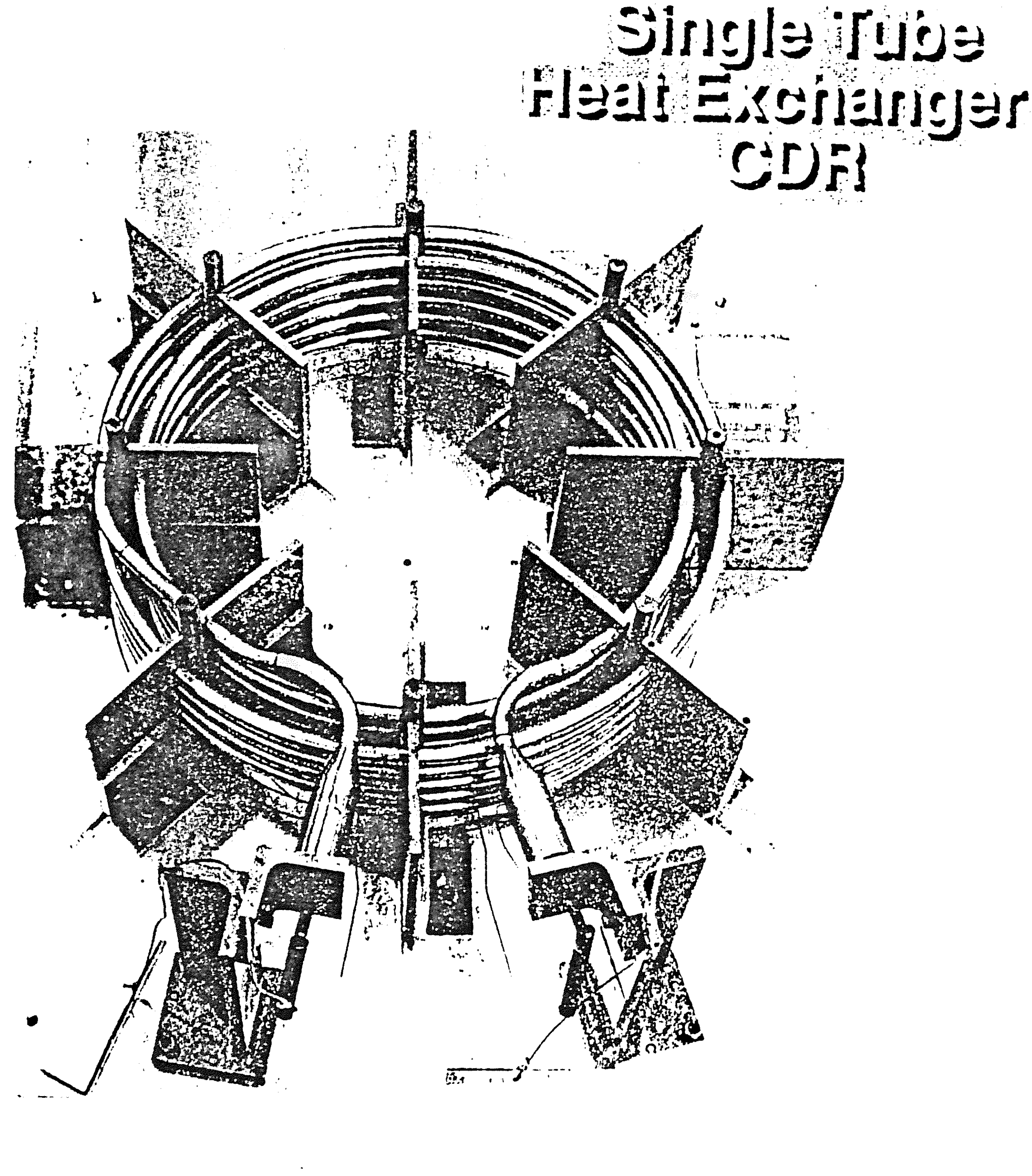


Figure 8

1.6 Phase II+ Powerhead

Function

The SSME powerhead is an assembly of eight major units i.e. the hot gas manifold which serves as a structural base for mounting the main injector, the two preburner injectors, the heat exchanger coil, the two high pressure turbopumps and the main combustion chamber. This creates a compact, efficient package where the pump turbines are very close coupled to the preburners, and duct lengths and losses are held to a minimum.

Comparisons

The objective of the Phase II+ Powerhead is to reduce the HPFT and hot gas manifold transverse delta p, reduce the hot gas flow resistance, improve the flow distribution through the fuel side transfer tubes, reduce the main injector oxidizer post high velocity hot gas impingement, reduce the dynamic stress in the preburner injector elements, reduce maintenance, and increase the operating margin at 109% rated power level. The design solutions incorporated to achieve the intended results include increasing the hot gas flow area, improving the hot gas wall contour, eliminating the center fuel transfer duct, shorten the preburner injector elements, and improve the main injector flow shield design. The powerhead components that are affected include the hot gas manifold, the preburners, the preburner fuel supply ducts, the coolant supply ducts, the main injector and the heat exchanger. The hot gas manifold and the heat exchanger modifications will be covered under the individual topic headings separate from the Powerhead discussion. Figure 9 shows the cutaway drawing of the Phase II+ powerhead including the HPOT and HPFT. The LOX preburner injector elements post length is reduced 0.490 inches which eliminates 5 rows of fuel sleeve holes. This reduces the element fuel sleeve resistance. See Figure 10. The fuel preburner injector is moved forward 1" to accomodate the 2 duct hot gas manifold. The main injector flow shield open area is increased to reduce the hot gas pressure to prevent hot gas ingestion in the coolant circuit which has been identified as a cause of MCC and baffle degradation. The LOX inlet elbow will be forged instead of cast to reduce the high incidence of defects (see Figure 11). The turbine drive low pressure fuel T/P duct is rerouted to clear the fuel preburner (see Figure 12). The rigid fuel bleed duct mounting bracket is revised to clear the HGM torus. In addition, several lines and brackets on the Powerhead are modified to accomodate these changes. Further detailed descriptions of the Powerhead modifications can be found in the SSME Powerhead CDR, 24 January 1991.

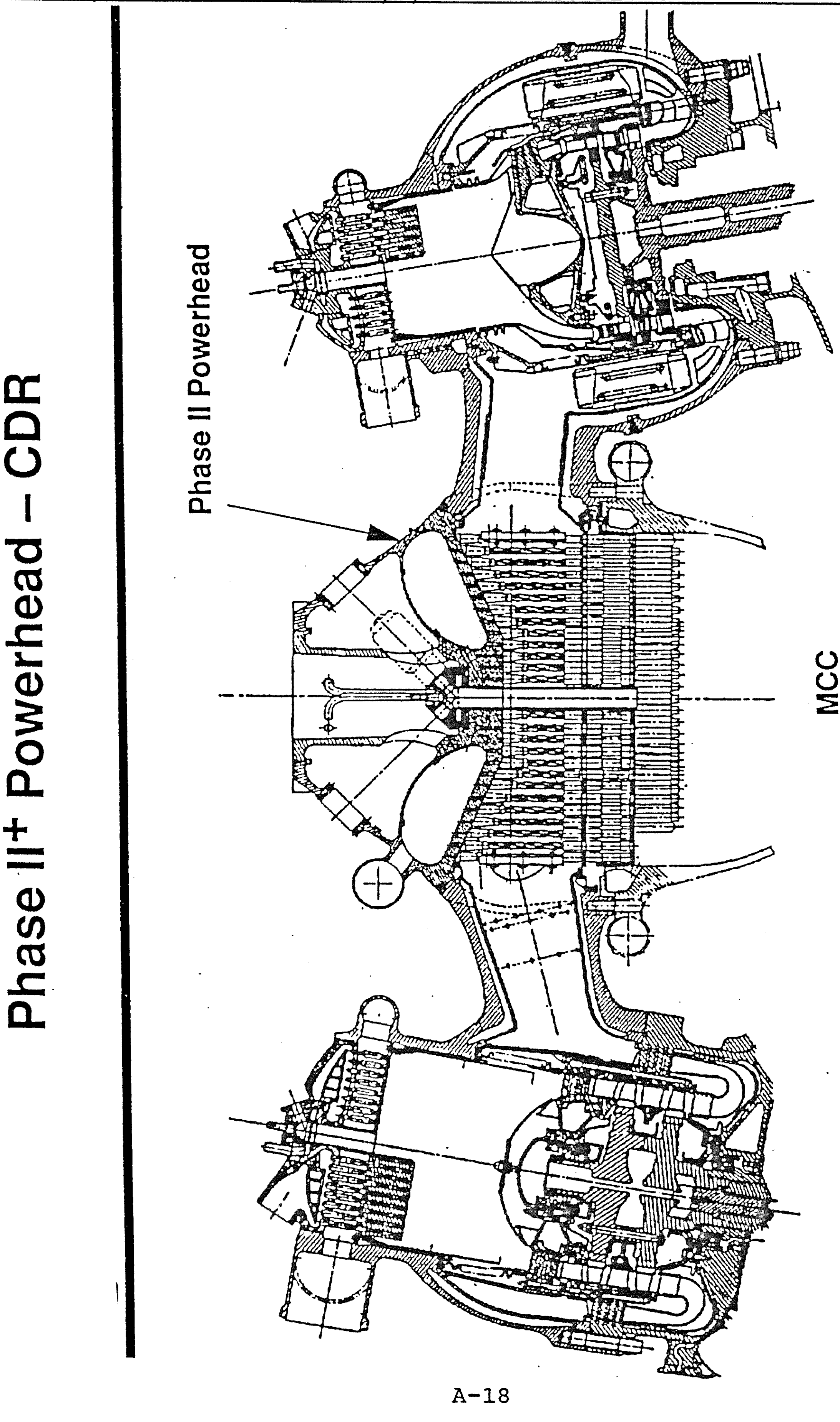
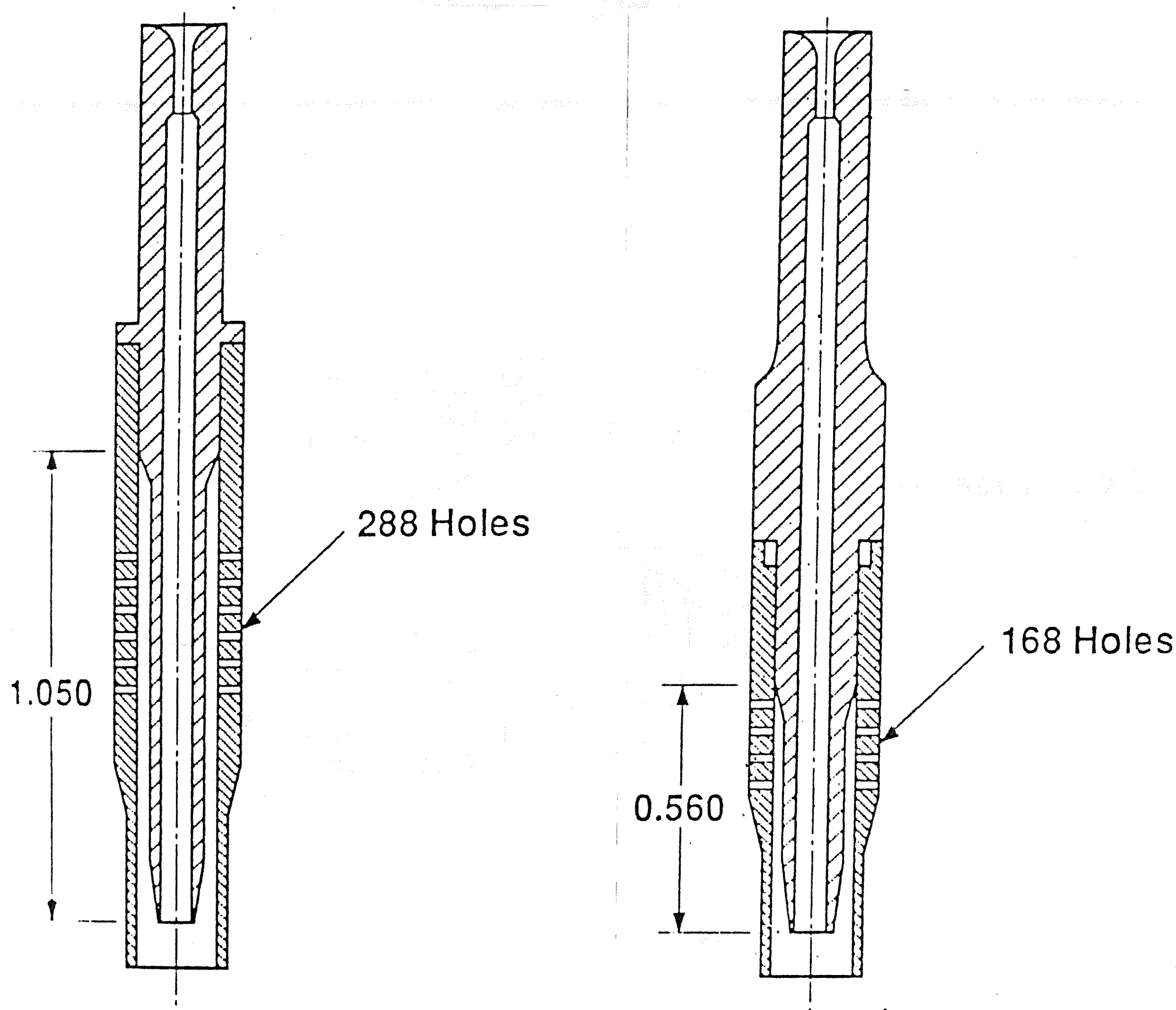

Phase II+ Powerhead – CDR
Phase II Powerhead
HPOTP
MCC
Phase II Engine
HPFTP

Figure 9

Elements - Injector



Phase II
(including LOX post support pins)

Phase II+

Figure 10

Phase II+ Powerhead CDR
Forged LOX Inlet Elbow

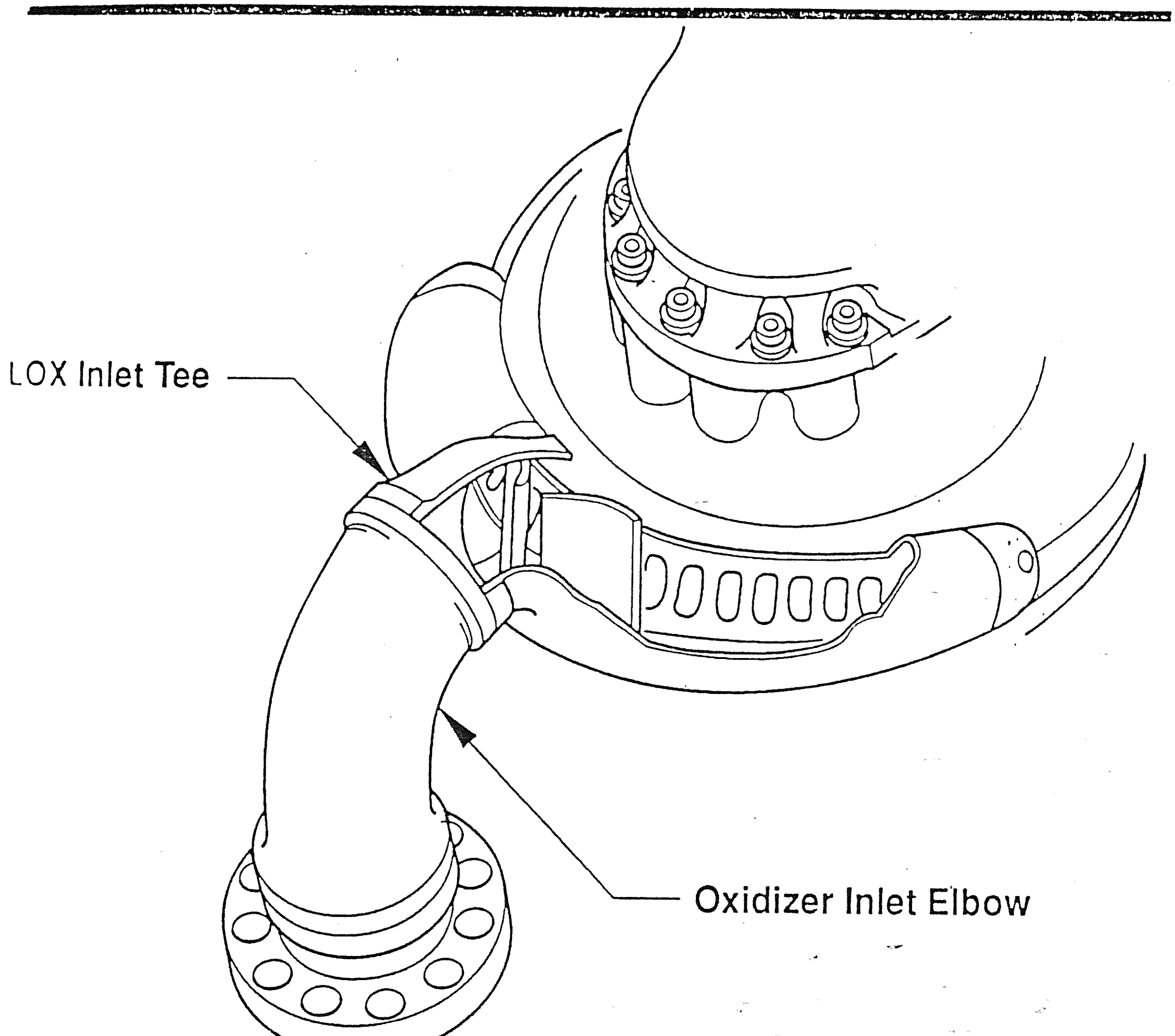


Figure 11

Phase II+ Critical Design Review Ducts

Engine Orientation

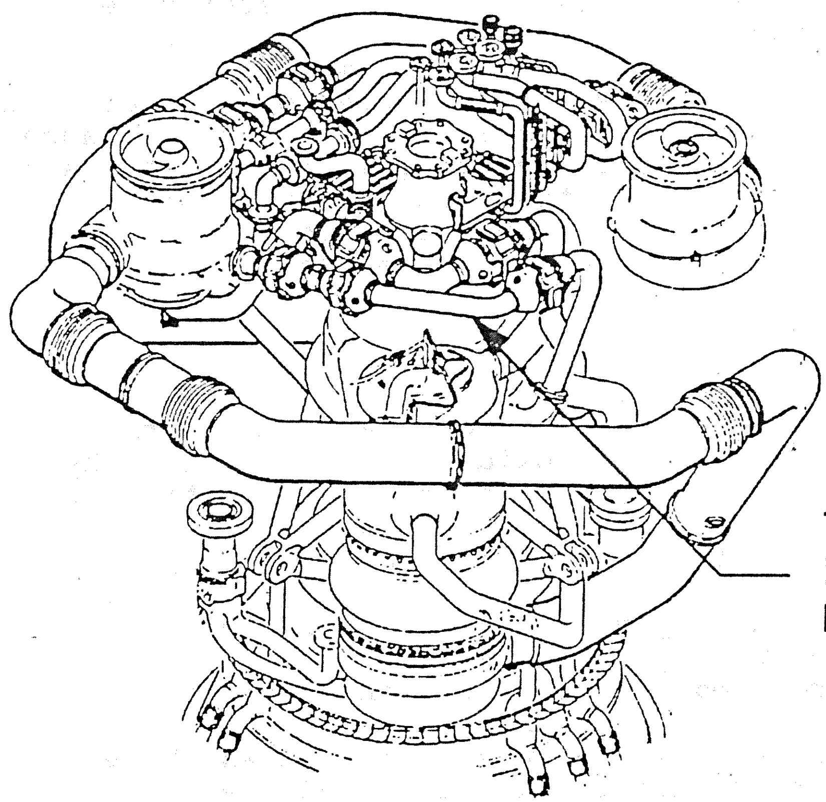


Figure 12

1.7 Two-Duct Hot Gas Manifold

Function

The hot gas manifold (HGM) conducts hydrogen rich steam from the turbines to the main injection elements of the main chamber injector. The area between the wall and the liner provides a coolant flow path for the hydrogen gas that exhausts from the low pressure fuel tubopump (LPFT) turbine, protecting the outer wall and liner against the temperature effects of the hot gas from the preburners. This hydrogen is the fuel from the injector baffle elements and also the coolant for the primary faceplate, the secondary face, and the main combustion chamber (MCC) acoustic cavities. The oxidizer side of the HGM has a canted flange to which the high pressure oxidizer turbopump (HPOT) is studmounted. Heat exchanger tube supports are welded to the inner wall on the oxidizer side of the HGM. Two hot gas transfer tubes route the HPOT turbine exhaust gas to the main injector torus manifold, where it is radially directed into the main injector. The oxidizer preburner is welded to the upper end of the oxidizer side of the HGM. The fuel side also has a canted flange to which the HPFT is studmounted. Three hot gas transfer tubes route the HPFT turbine exhaust gas to the main injector torus manifold, where it is radially directed into the hot gas cavity of the main injector. The fuel preburner is welded to the upper end of the fuel side of the HGM.

Comparisons

The hardware affected on the hot gas manifold by the Phase II+ changes include the fuel and oxidizer half shells, the hot gas manifold (HGM) liners, and the upper and lower transition rings. The fuel collector bowl area was increased which reduced the transverse delta p, flow velocity, and flow resistance (see Figure 13). The faired inlet was changed to a smooth flow path which reduces the turbulence and flow resistance. The fuel and oxidizer transfer tubes flow area was increased which reduced the flow velocity, reduced the drag load on the main injector LOX posts, and improved the flow uniformity (see Figure 14). The main injector torus flow area was increased which reduced the circumferential flow velocity and improved the circumferential pressure distribution (see Figure 15). Two constant cross section elliptical transfer tubes replace the three tubes currently used (see Figure 16). The structure was reinforced to maintain the elliptical shape under pressure. The horizontal alignment provides maximum flow area at the torus entrance. The advantages of the two duct hot gas manifold over the current three duct is that the design reduces flow velocity and fluctuating pressures (75% at the main injector). The turbine transverse delta P is reduced i.e. 60% at the HPFT turbine exit pressure, and maintenance is reduced i.e. no preburner support pins, 47 welds are eliminated and 903 overlays are eliminated.

Phase II+ Powerhead – CDR

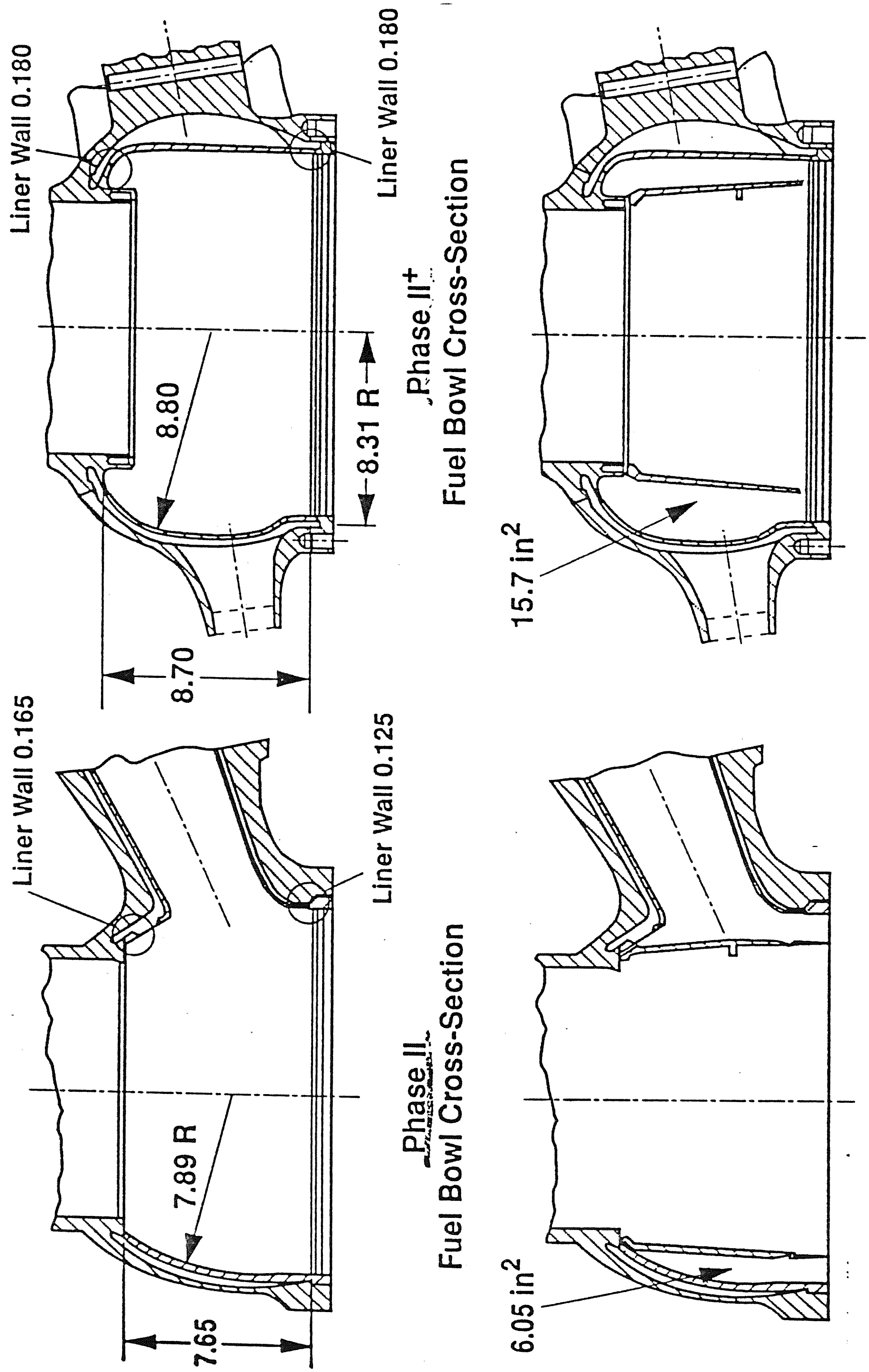


Figure 13

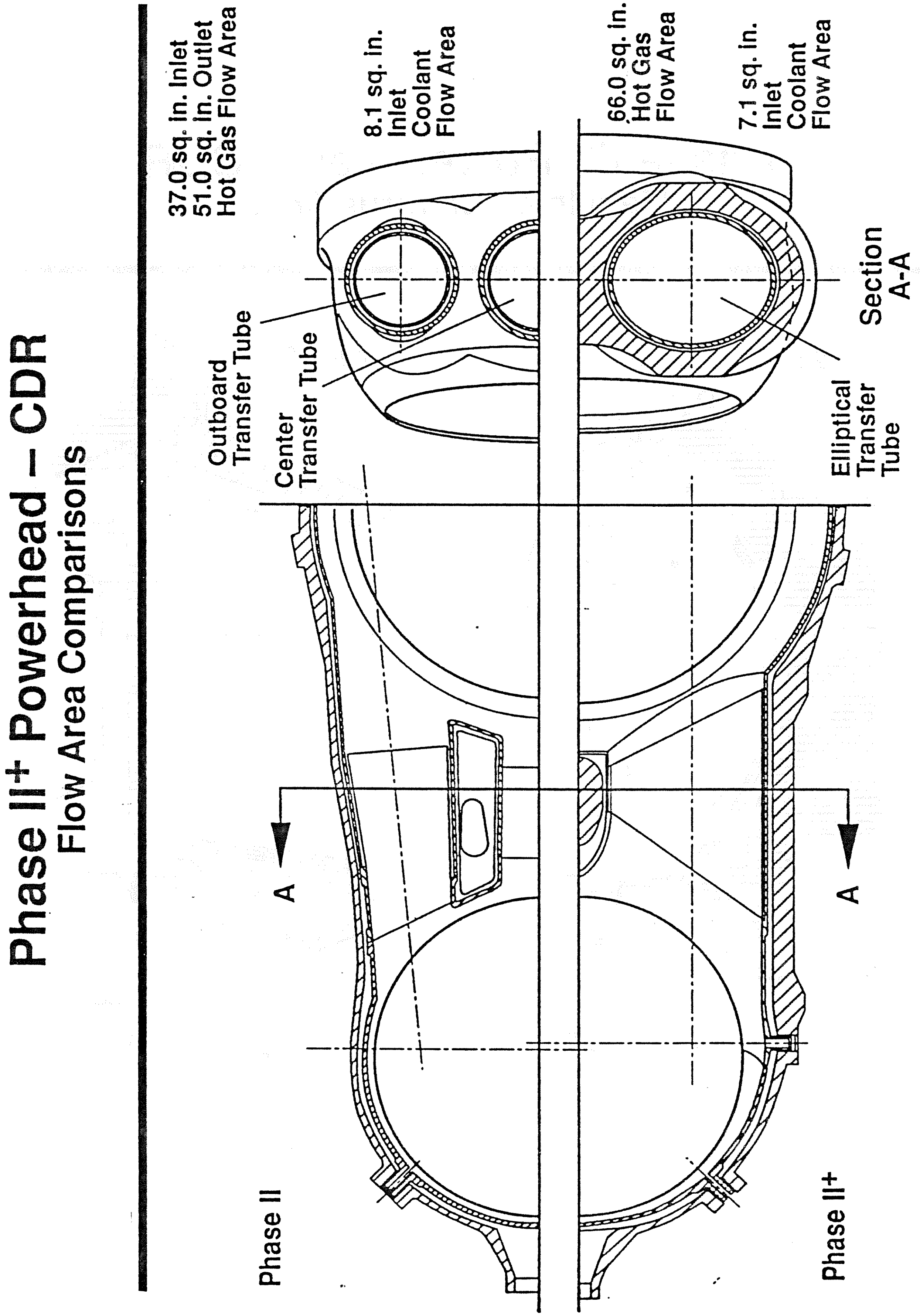

Phase II+ Powerhead – CDR
Flow Area Comparisons
37.0 sq. in. Inlet
51.0 sq. in. Outlet
Hot Gas Flow Area
8.1 sq. in.
Inlet
Coolant
Flow Area
66.0 sq. in.
Hot Gas
Flow Area
7.1 sq. in.
Inlet
Coolant
Flow Area
Outboard
Transfer Tube
Center
Transfer Tube
Elliptical
Transfer
Tube
Section
A-A
A
A
Phase II
Phase II+

Figure 14

Phase II+ Powerhead – CDR
Main Injector Torus Area

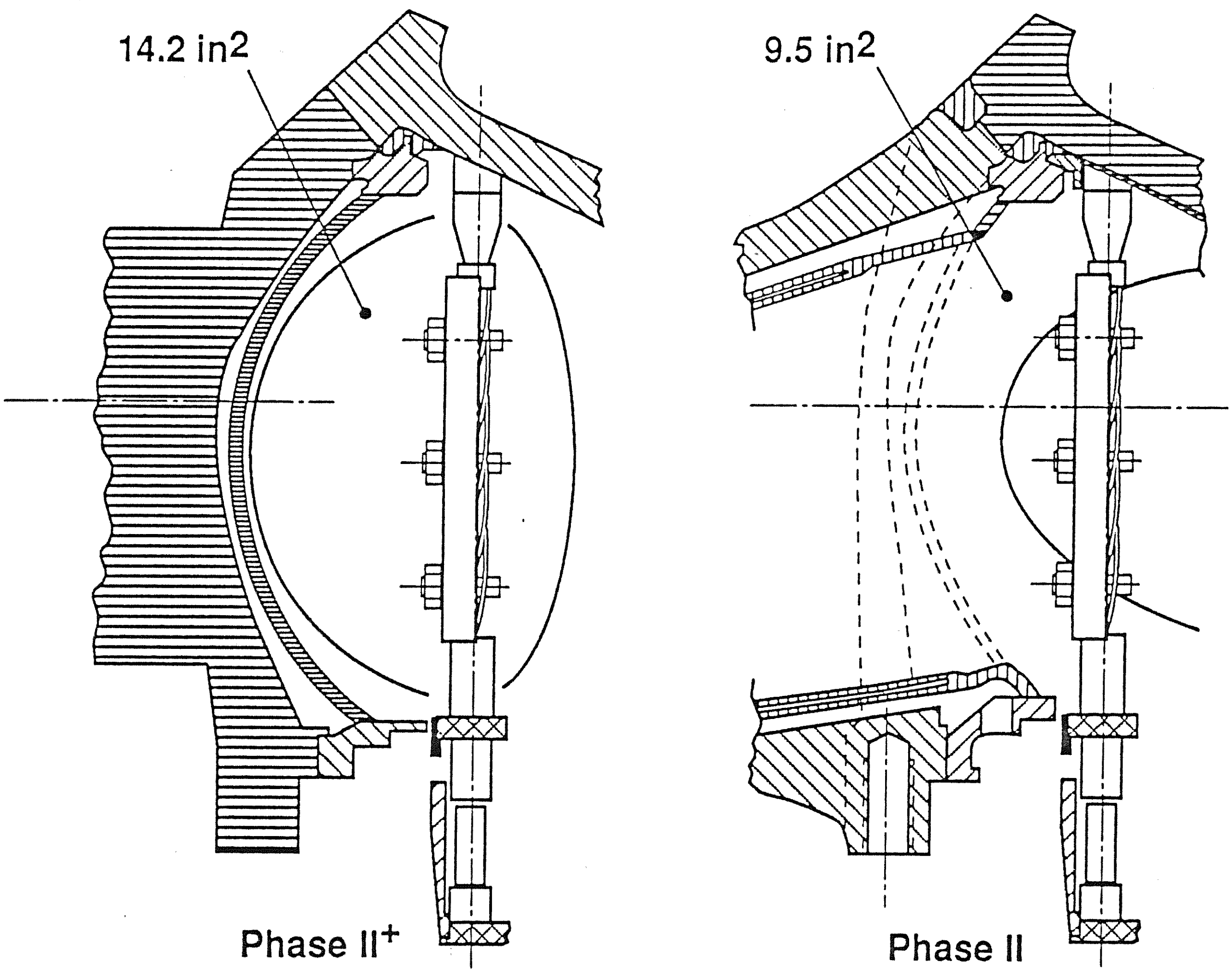


Figure 15

Phase II$^+$ Powerhead – CDR
HGM Comparison

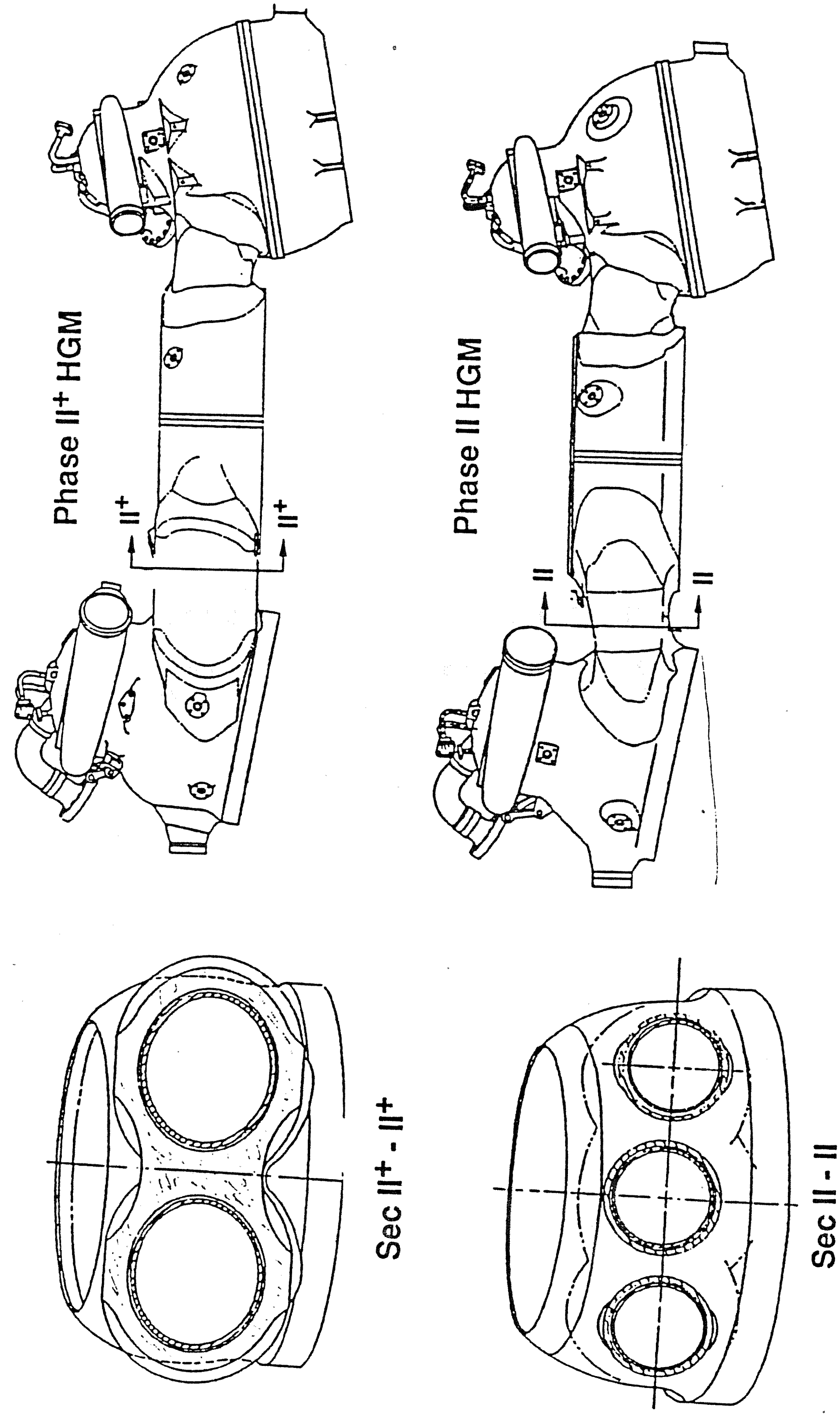


Figure 16

SECTION 2: SOLID ROCKET MOTORS

2.1 Advanced Solid Rocket Motor
2.2 Redesigned Solid Rocket Motor Igniter

2.1 Advanced Solid Rocket Motor (ASRM)

Function

The solid rocket motors (SRM)are the primary propulsive element providing impulse and thrust vector control from ignition to solid rocket booster (SRB) staging. The SRM consists of a lined, insulated, segmented rocket motor case loaded with solid propellant; an ignition system, initiators, and loaded igniter; a movable nozzle; systems tunnel bracketry; instrumentation; and the integration hardware. The purpose of the advanced solid rocket motor (ASRM) is to provide a safer higher performance booster for the STS program.

Comparisons

The advanced solid rocket motors are jointly manufactured by Lockheed, Aerojet, Thiokol and Rust. Lockheed will provide project management, support equipment, quality assurance and facilities management. Aerojet will provide engineering manufacturning, quality assurance, and testing. Thiokol will manufacture the nozzle, and Rust will provide architecture, engineering and construction. The current timeline shows the first flight to be Fall 1996. The telemetry downlink is similar to the SRB's with a large reduction or elimination of the throttle bucket. This will simplify flight control ascent operations and removes several SSME failure modes. The lower temperature qualified polymer seals will require less temperature dependent launch committ criteria (LCC) constraints than the present SRB. It will also involve the elimination of several electrical strip heaters and heated gaseous purges. The new design is being developed to minimize changes in other orbiter/ET hardware as well as flight software and operations. The lift-off weight for the redesigned solid rocket motor (RSRM) is 1,256,000 while the ASRM is 1,348,000 lbs. The motor mass fraction of the weight for the RSRM is .882 and for the ASRM is 0.894. The ISP at vacuum for the RSRM is 267.9 seconds compared to 270.3 sec for the ASRM. The average thrust for the RSRM is 2,590,000 lbs and for the ASRM is 2,624,000 lbs. The length in inches of the RSRM is 1513.3 the same as the ASRM. The width of the RSRM is 146.0 inches while the ASRM is 150.0 inches. The case factory joints of the RSRM are tang/clevis /pin. The ASRM are welded. The case field joints of the RSRM are tang/clevis/pin (3 per RSRM). The ASRM are bolted (2 per ASRM). The thrust vector control of both are flexible bearings. Both solid rocket motors are recovered and reused. The ASRM improves ascent performance over the RSRM by 6000 lbs per SRM or 12,000 lbs total. SSME throttling at max Q is minimal for the ASRM. The nominal separation time for the RSRM is 2:06 compared to 2:16 for the ASRM. The first stage abort options will not change for the ASRM. The abort boundary windows will improve with the ASRM giving larger windows. Mixed

ASRM/RSRM flight operations will begin Fall 1996. Some major program concerns include the costs factor of the ASRM compared to the RSRM, and the availability of the ASRM to support space station hardware launches. The advantages of the ASRM's include throttle bucket reduction or elimination, wider abort boundary regions, better case field joints, lower temperature qualified polymer seals, fewer potential hot gas leak paths i.e. 30 vs 258 critical leak paths, and higher levels of automated robotic process control and manufacturing. The disadvantages include an unstable propellant, higher thermal and stress loads in ascent, and higher reentry thermal and stress loads. The ASRM will have higher performance - 12,000lbs extra payload or equivalent amount of performance translated to extra velocity or higher inclination if the payload capacity does not change. The specific impulse is greater by 2.5 sec. There is a different thrust versus time profile for the ASRM's. The thrust of the ASRM compared to the present SRM will be higher at T-0, drop lower in the max Q region, and burn longer before separation. SRB SEP will move from 2:06 to 2:16 under nominal conditions. Lower ASRM thrust from T+40 to T+80 will mean the SSME's no longer need to throttle deeply during max Q. Present projections call for the SSME's to stay at 100% power from T-0 to T+75 and then throttle up to 104% for the remainder of ascent to 3-g throttle back just prior to MECO. SEP will be 30,000 feet higher altitude under nominal separation conditions, 180,000 vs 150,000. A nominal MECO may occur earlier than 8:31. Present plans call for flights the first few years of ASRM operations to include switching ASRM's and the present SRM's flight sets back and forth depending on mission performance capabilities needed, inventory in stock, and ASRM production rate. Eliminating the SSME throttle bucket eliminates 175 failure modes for the shuttle. The continuous propellant mix process uses constant sampling by the Fourier Transform Infrared technique. It is expected to reduce propellant production costs by 2/3 over the batch method. (HTPB) hydroxy terminated polybutadiene propellant is slightly more energetic than the polybutadiene acrylic acid acryinitrile terpolymer. Stronger binder allows more of the energetic components in the mix. The motor case is welded factory joints and integral external tank atachment rings which eliminate failure modes. The ASRM integral stiffeners, 99ni-4Co-0.3C steel and 2267 fewer parts reduce the case weight, splashdown damage and refurbishment costs. Nozzle-omniaxis moveable, carbon-carbon integral throat nozzle eliminates 72 seals and two of five internal joints. The igniter-cartidge loaded HTPB igniter reduces leak paths and offers tailored soft ignition. The PDR is set for FEB 1992, and the CDR is set for August 1994.

2.2 RSRM Igniter Seal Redesign

Function

The solid rocket motor igniter is a forward end, internally-mounted, solid-rocket type igniter, an igniter initiator, an igniter adapter, and a safe and arm device. It contains 191 lb of propellant which is 69% ammonium perchlorate, 10 percent spherical aluminum, 3% ferric oxide, 18% P BAN polymer and epoxy resin. The igniter system initiates SRB firing.

Purpose

The purpose is to prevent hot gas blow by which is potentially catastrophic.

Comparisons

The igniter redesign development and qualification will be complete to support the first flight on RSRM-29 (STS-54). The igniter redesign implementation is to be accomplished in two phases. The structural elements are to be tested in TEM-8, and then the interim design is to be implemented on STS-45 with putty filler in the joint. The all up design with the J-leg is to be tested in TEM-9 and implemented on STS-54. The redesigned thicker adapter is designed to reduce the seal gap opening during pressurization. Strainsert bolts are used to preload the joint at lower specification limits for best verification. One-sided putty layup at the igniter outer joint is used to decrease propensity for air entrapment and potential blow paths. The interim design test was conducted July 31, 1991. Preliminary data indicates nominal functioning. The igniter redesign is to be tested MAR 1992. Figure 17 shows the redesigned igniter with the thicker adapter, strainsert bolts, and putty location.

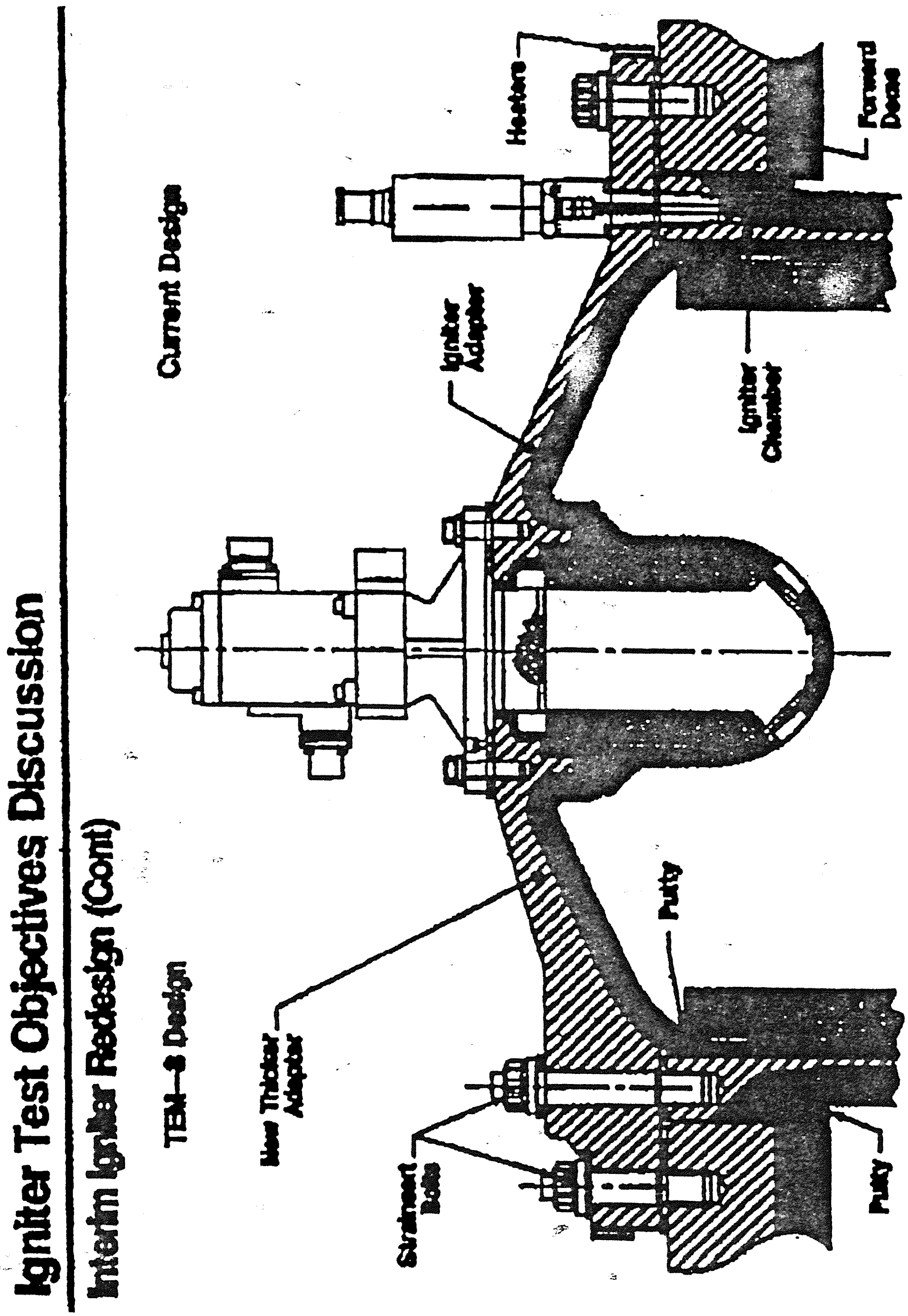

Igniter Test Objectives Discussion
Interim Igniter Redesign (Cont)
TEM-6 Design
Current Design
Heaters
Forward Dome
Igniter Adapter
Igniter Chamber
Putty
New Thicker Adapter
Strainsert Bolts
Putty

Figure 17

SECTION 3: MAIN PROPULSION SYSTEM

3.1 -006 Helium Pressure Regulator
3.2 GOX Fixed Orifice
3.3 GH_2 Fixed Orifice
3.4 14 Inch Disconnect

3.1 -006 750 Helium Pressure regulator

Function

The Helium regulator regulates the engine helium supply pressure from 4500 to 900 psig down to 750 psig, the nominal operating pressure at the orbiter/engine interface. A relief valve will prevent downstream overpressurization should the regulator fail high. A redundant regulator is in a parallel leg. One regulator is used in the orbiter pneumatic system to regulate the high pressure helium supply to a lower pressure for use in component operation, propellant feed and fill systems, pressurization to expel propellants, and to provide purge gas to the engines.

Comparison

The primary purpose for the redesign was to preclude a failed open regulator from a dynamic fatigue failure of the regulator sensing bellows. This failure mode was demonstrated on the -004 regulator which is similar in design to the -005. The -006 design is the result of knowledge gained over the last three years. The -006 design eliminates unpredictable flow passages, adds seals and orifice control of areas such as the mainstage and controller assembly, eliminates variable damper clearance for better response capability, isolates the bellows from noise, and vents the bellows. The design stability characteristics were shown to be stable by the dynamic regulator model. The -006 configuration including the housing, bellows capsule, poppet/seat/retainer, and belleville supports designs are identical to the -005. Figure 18 shows the -005 configuration, and Figure 19 shows the new -006 design. The flow passages are similar, and the internal seals are similar or identical. The -006 regulator has three major new features which distinguish it from the -005 regulator. The -006 regulator has a vented sensing bellows, whereas the -005 regulator has a bellows pressurized with 70 psia of freon. The vented bellows design changes the regulator from an absolute regulator to a gage regulator, or the regulated pressure supplied by the -006 regulator will depend on the ambient pressure. The -005 regulator was independent of ambient pressure. For a given inlet pressure and flow rate, the regulator pressure will drop 14.7 psia from sea level to vacuum operation for the -006. The -005 was thermally compensated using freon to cool which softened the bellow and countered the increased stiffness in the belleville springs. The -006 regulator will not have thermal compensation. The -005 regulator used elgiloy steel which has excellent mechanical properties across a wide range of temperatures, however the material was difficult to manufacture. The spring material was changed to maraging steel, which has similar mechanical properties, however the modulus of elasticity does change over the range of temperatures, It has been demonstrated that the -006 regulator band will drift 1 psi per every 10°F change in valve body temp, from the design reference of 70°F. If the valve is heated 10°F, the set point of the regulator will drop 1 psi. If the valve is cooled 10°F, the set point will rise 1.0 psi. For a given inlet pressure, the -005 regulator pressure would drop as the flow rate increased. For a given flow rate, the regulator pressure will be relatively constant for an inlet press of 4500 to 3000 psia. The regulator pressure would then gradually rise as the inlet pressure decreases until a critical pressure is reached. For inlet pressures below the critical pressure, the regulator pressure would always decrease. In contrast, for a given inlet pressure the -006 regulator outlet pressure can increase as the flow rate increases, which is the exact opposite behavior of the -005 regulator. This only occurs with high inlet pressures. As the inlet pressure drops < 3000 psia, the -006 regulator begins to behave like the -005 regulators. First flight of the -006 for each orbiter is as follows: STS-47 OV-104, STS-48 OV-103, STS-50 OV-102, OV-105 will fly with the -006 on its first flight.

COMPONENT: REGULATOR, HELIUM PRESSURE, 750 PSIG

Dash -0005

MATERIALS
HOUSING: 21-6-9 CRES MAIN POPPET: 21-6-9 PILOT POPPET: VESPEL SP-21 PISTON: 6061-T651 AL ALLOY BELLOWS: INCONEL 718 BELLEVILLE: ELGILOY

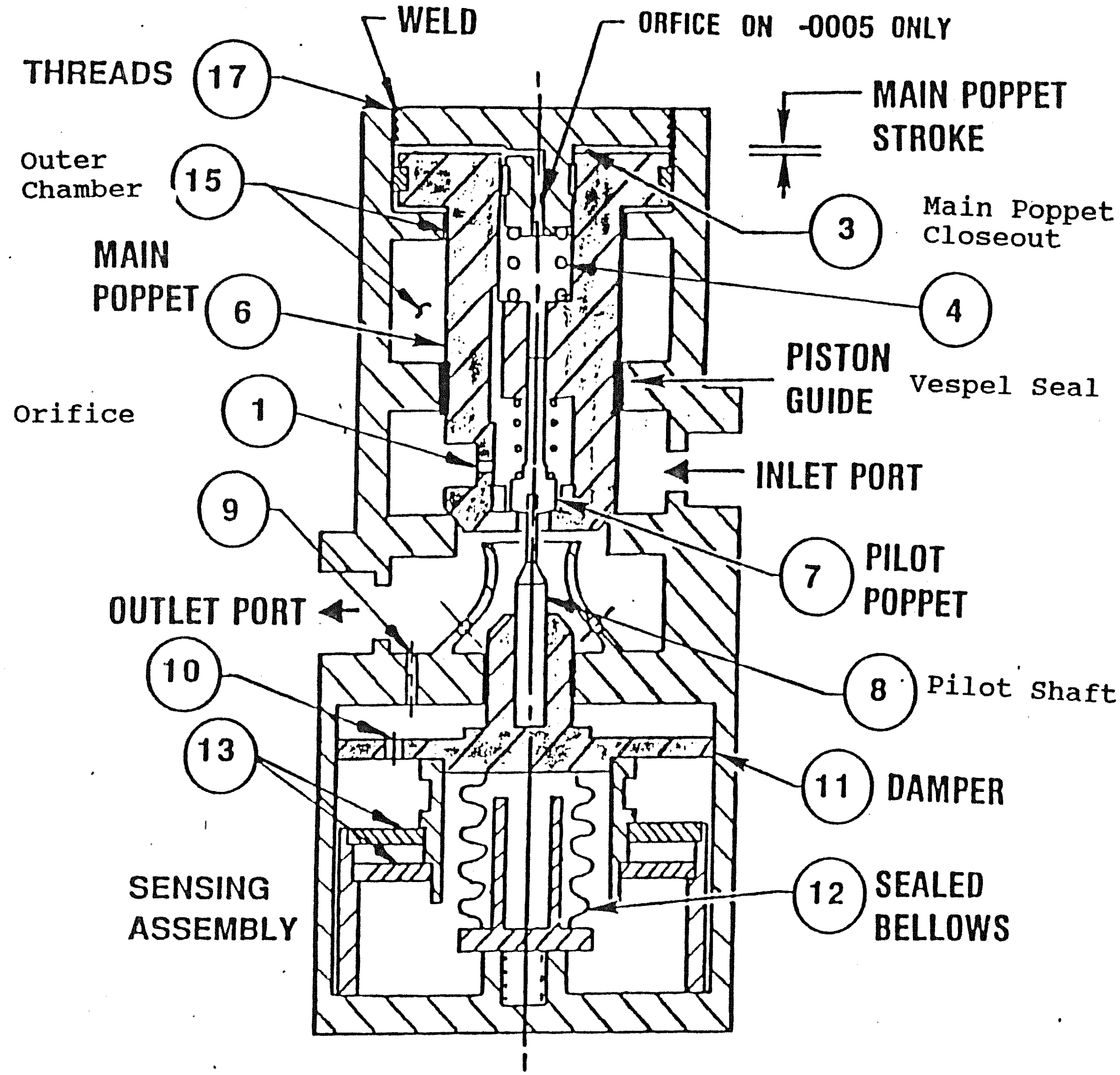


Figure 18

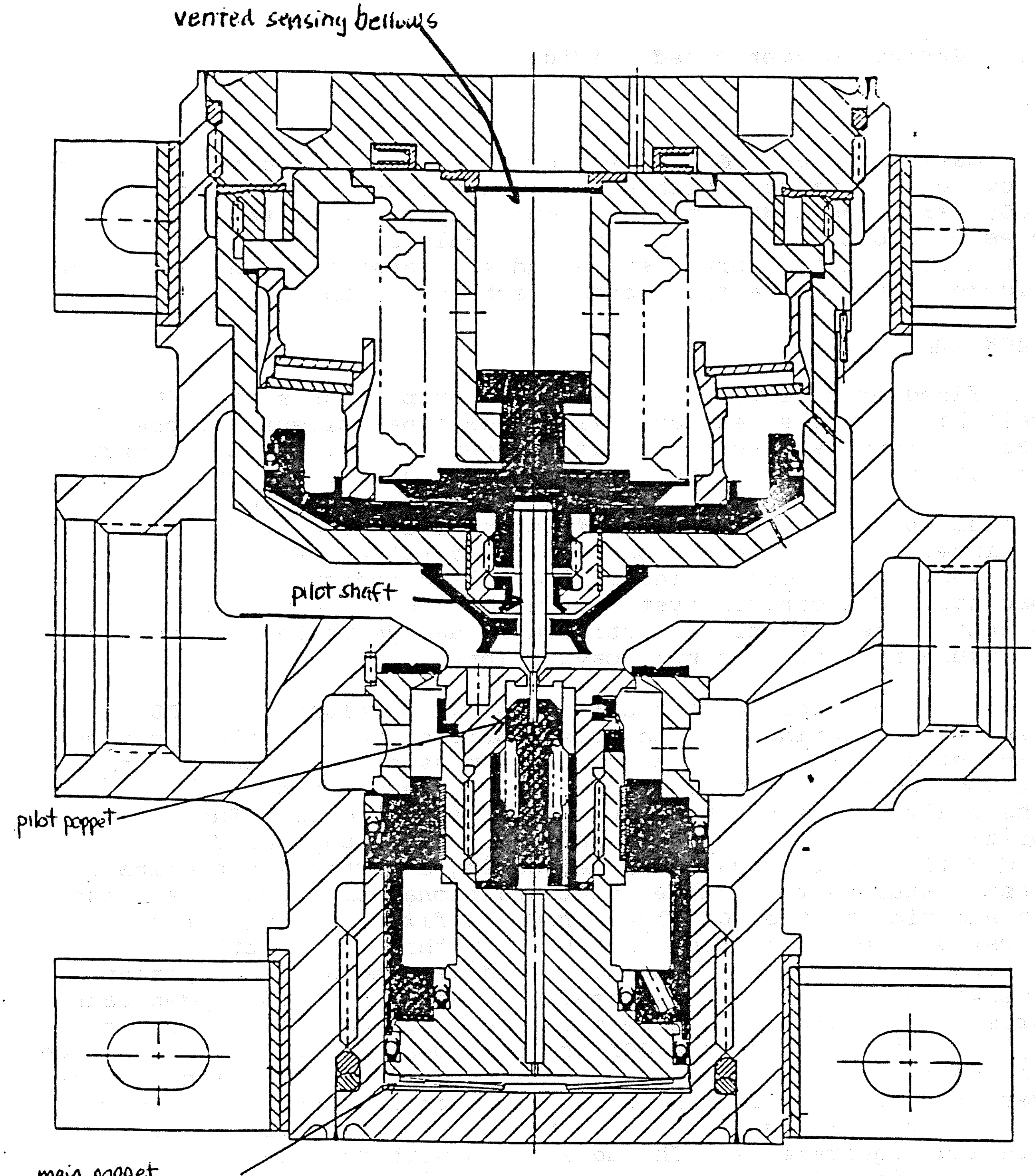


Figure 19

750 PSIG HELIUM REGULATOR 73664-0006 CONFIGURATION

3.2 Gaseous Oxygen Fixed Orifice

Function

The gaseous oxygen (GOX) fixed orifice is designed to replace the flow control valves which supply pressurant i.e. gaseous oxygen (GO_2) from the SSME's to the liquid oxygen (LO_2) tank for pressure control. The flow control valves (FCV's) provide 100% rated flow in the normal state and 42% rated flow when activated. Figure 20 shows the flow control scheme for the GOX.

Background

The fixed orifice will increase main propulsion system (MPS) reliability and safety and will allow final closure of open failure reports associated with flight anomalies such as particle impact sensitivity. The current system utilizes center, left, and right space shuttle main engine (SSME) GOX flow control valves where each have a bypass channel and a control channel equipped with a normally open solenoid valve. Each control valve is electrically controlled by the ullage pressure control package. The control system is needed to prevent both overpressure resulting in structural damage to the ET and under pressure resulting in pump cavitation.

The fixed orifice for the GOX system was evaluated in 1980 with results indicating that the fixed orifice would violate minimum tank structural limits early in the mission or cause O_2 venting around the time of solid rocket booster separation (SRB SEP). The analysis was not based on actual flight data. The fixed orifice was evaluated again in 1986. The study was dropped until a GOX flow control valve failed during a particle contamination test. Studies then showed that additional pressurant is needed in addition to the GOX flow through a fixed orifice during the first 50 seconds of boost and between three G throttling and main engine cut-off (MECO). Helium could be used as a supplement. Results of subsequent tests showed that the liquid oxygen tank pressure requirement can be satisfied with one set of GOX and helium orifices with the GOX orifice equal to 0.0204 in^2/SSME and the Helium orifice equal to 0.0174 in^2. The conclusions reached were that the fixed orifice GOX flow control concept can be made to work with Helium (He) augmentation for tank structural and venting requirements. The advantages with the fixed orifice are a lower risk of fire hazard due to particle contamination, positive pressure control due to active helium flow control, minimized uncertainties in ullage heat transfer mixing, increased flexibility, and a net payload gain of 115 lb. In 1988 new tank structural limits allowed wider ullage pressure excursions thereby making the fixed orifice feasible without He augmentation.

Verification of the fixed orifice pressurization system is accomplished in three steps as follows:

- o Step 1: FCV's shimmed to 93% high/55% low (FCV's active) STS-41
- o Step 2: FCV's shimmed to 85% high/66% low (FCV's active) STS-38
- o Step 3: FCV's shimmed to 78% (FCV's disabled-first fixed orifice flight)

STS-41 and STS-39 were the flights with the Step 1 configuration. All GO_2 FCV's operated nominally. STS-38 and STS-37 were the next launches with the step 2 configuration, and STS-40 flew the 78% shimmed valves. Note: Use of the fixed orifice with the Advanced Solid Rocket Motor (ASRM) may not be feasible. In first stage there are minimal structural constraints because of dynamic loads and high Q bar on the nose as the LOX tank is the structural support). The fixed orifice will eliminate three flow control valves which will result in a decrease of weight from hardware of 12.6 lbs and pressurant mass of 128 lbs. It is currently planned to fly the 78% shimmed valves for some time to come. A decision is expected Fall 1993 whether to go to a fixed orifice as a final solution or remain with the shimmed valve.

GO_2 PRESSURIZATION SYSTEM SCHEMATIC

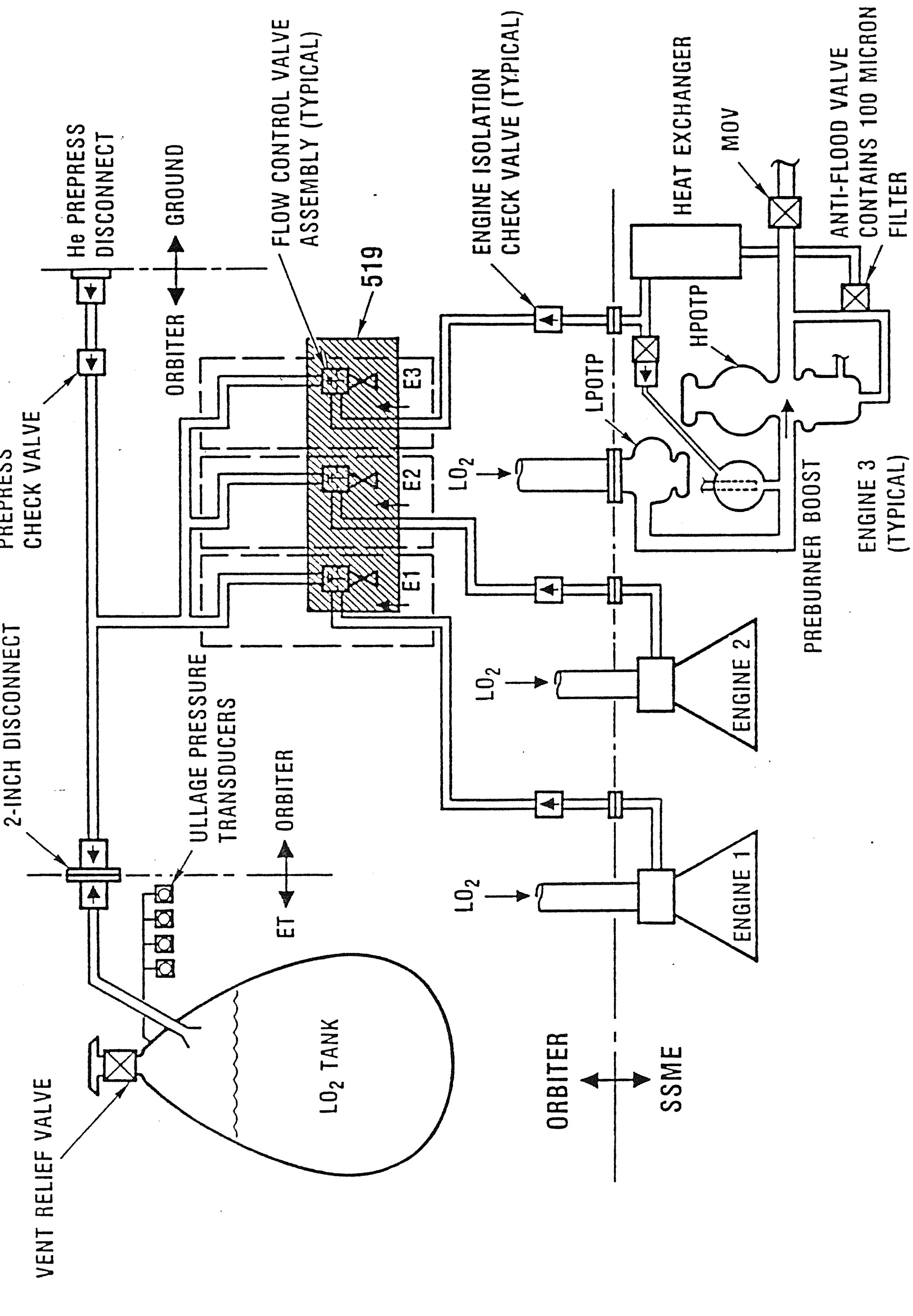


Figure 20

3.3 Hydrogen Fixed Orifice

Function

The Fixed Orifice is designed to replace the flow control valves which are used to control the hydrogen pressurant flow from the three main engines to the external tank. There are a total of three valves, one per main engine. Each valve provides independent control of the pressurant delivered by the respective engine.

Background

The fixed orifice reduces crit-1 failures resulting from multiple flow control valve (FCV) failures i.e. tank structural failure for 2 or 3 FCV's failed closed and H_2 venting and burning if 2 or 3 FCV's failed open. A precedence was established on STS-28R FCV no.1 did not operate normally from T+7 to MECO. The anomoly was due to contamination between the sleeve and moving part. An analytical model was developed to accurately predict fixed orifice pressurization characteristics. Maximum ullage pressure is required at start of 3g throttling. Ullage pressure requirements defined for the fixed orifice pressurization study are based on dispersions in feed system pressure loss, supply temperature, and 3g throttling. The study showed that max/min fixed orifice operating band is greater than presently constrained by the ET and SSME's. The GH_2 fixed orifice pressurization system cannot satsfy both present tank and SSME net positive suction pressure (NPSP) requirements. If the 14 inch disconnect is installed, the line pressure loss will be less, therefore the pressure at the suction of the pump will be greater. The SSME's could be throttled just prior to 3g throttling where the NPSP requirements are at a max. The performance penalty is small i.e. (100 lb of thrust) and is compensated for by lower ullage mass. GH_2 fixed orifice requirements can be satisfied with installation of the 14 inch disconnect and minor change to tank and SSME requirements. Crit 1 failures due to multiple GH_2 FCV failures may be eliminated by the fixed orifice. Conclusions are that the fixed orifice system cannot satisfy present tank and SSME requirements but is possible if the maximum pressure limit is increased by 1.6 psi, and the SSME NPSP requirements are reduced by use of the 14 inch disconnect or by throttling the SSME's at 400 sec. If the fixed orifice concept is not feasible, it has been recommended that a more failure tolerant FCV shim schedule be implemented i.e. change the FCV open/closed flow areas to 80/40%

Current Status as of July 19,1991

The current SSME inlet pressure and/or ET LH_2 upper structural limits may be violated if the fixed orifice is implememnted.

This is based on revised dispersion assessments and if nominal +/- 3 sigma performance is maintained within limits. Further shimming the FCV from 100/18% to 51/46% can improve present pressurization systems failure tolerance. Present ullage pressure requirements can be maintained with 3 FCV's failed closed. If 2 FCV's fail open the dispersed performance marginally violates the upper ET design limits. If the LH_2 tank relief setting is raised from 36 +/- 1 psi to 38 +/- 1 psi, this would provide 3 FCV fail open protection. The assessment of the revised FCV shim schedule is dependent on the fixed orifice assessment which is dependent on the feasibility of revising the normal trajectory throttle schedule.

3.4 14 Inch Disconnect

Function

The external tank (ET)/Orbiter feedline disconnect provides a means of loading and detanking the ET and supplying propellants to the orbiter main propulsion system (MPS). Each orbiter and ET disconnect half contains a pneumatically actuated flapper closure device which remains in its last actuated position (bistable). Prior to the start of tanking, the disconnects are opened and remain open until main engine shutdown for uninterrupted propellant flow. The valves are pneumatically closed after MECO and before tank separation to prevent propulsive venting of propellants which could cause ET/orbiter recontact, loss of manifold repressurization Helium supply during reentry, leakage of H_2 and O_2 into the engine compartment during TAL and RTLS aborts, and system contamination during entry.

Comparisons

The 14 inch disconnect uses swing flapper closure elements. In the open position, the flappers are moved out of the flow stream and pulled down into a protective cavity and kept there with mechanically induced hold down forces. The disconnect is designed to fit in the existing orbiter and external tank envelopes, the swing flapper disconnect provides identical attach points and equal or superior load carrying capabilities to the present 17 inch disconnects. The new units present signifiant advances in safety in the form of a verifiable locked open out of the flowstream closure element. Figure 21 shows the 14" disconnect in the open position. Figure 22 is the drivetrain in the open position. Figure 23 shows the disconnect in the closed position, and Figure 24 is the closed postion for the drivetrain. A 50% reduction in parts will improve the reliability. No field adjustments will be required with all critical motions and adjustments being inherent in the design and manufacturing and assembly process. Basically, the design provides a simple rugged mechanism which can perform all of the required functions such as a redundant means of holding the valves open during flow, a redundant means of driving the valves closed during separation, the surest possible closure for providing extremely low propellant leakage at cryogenic temperature, position feedback to verify the valves are open and locked prior to propellant loading and launch and closed prior to ET SEP and, no rigging or adjustment in the field. Test results from a cryo model and 14 inch disconnect flow model showed that the pressure drop was less than 5.4 psid, the hold open forces were 188 lbs_f for the ET and 242 lbs_f for the orbiter. There was no visible oscillation with the flapper and no effect on the engine cutoff sensors.

Differences between 17" and 14"

Item	17"	14"
Actuation	pneumatic piston	pneumatic piston

Item	17"	14"
Valve type	flapper	flapper
Back up closure if pneumatics fail	mechanical drive train at ET SEP	mech drive train at ET SEP
Valve position	in propellant flow	out of propellant flow
KSC field rig and adjustment	yes	no
Tight manufacturing tolerances	yes	yes
Redundant mechanisms to keep valve open during flow	yes	yes
	locked over center linkage mechanism	drive roller engagement with scroll cam detent
	mechanical latch	ET bellcrank roller engagement with orbiter fork cam detent
	fluid flow forces tend to open valve	flow tends to keep valve open
Position indicator	4 independent sets	3 independent sets
External leakage	triple redundant radial, rotary seals	triple redundant radial, rotary seals
LO_2 delta p (3 at 109%)	3.7 psi	1.6 psi
NPSP	adequate	higher margins during nominal runs lower NPSP margins at MECO
Engine cutoff sensor response	adequate	timing change required

Propellant residual	adequate	no sig change
Individual parts	345	166

Item	17"	14"
Flow area	adequate	less flow area results in less chance of ET recontact if disconnect valve fails open at SEP for RTLS and TAL
Weight	288 lbs	288 lbs
Testing	extensive	limited - cryo flow and water flow tests no Lh_2 flow tests
First Flight	current	Feb 1993

PROGRAM CANCELED IN 1992

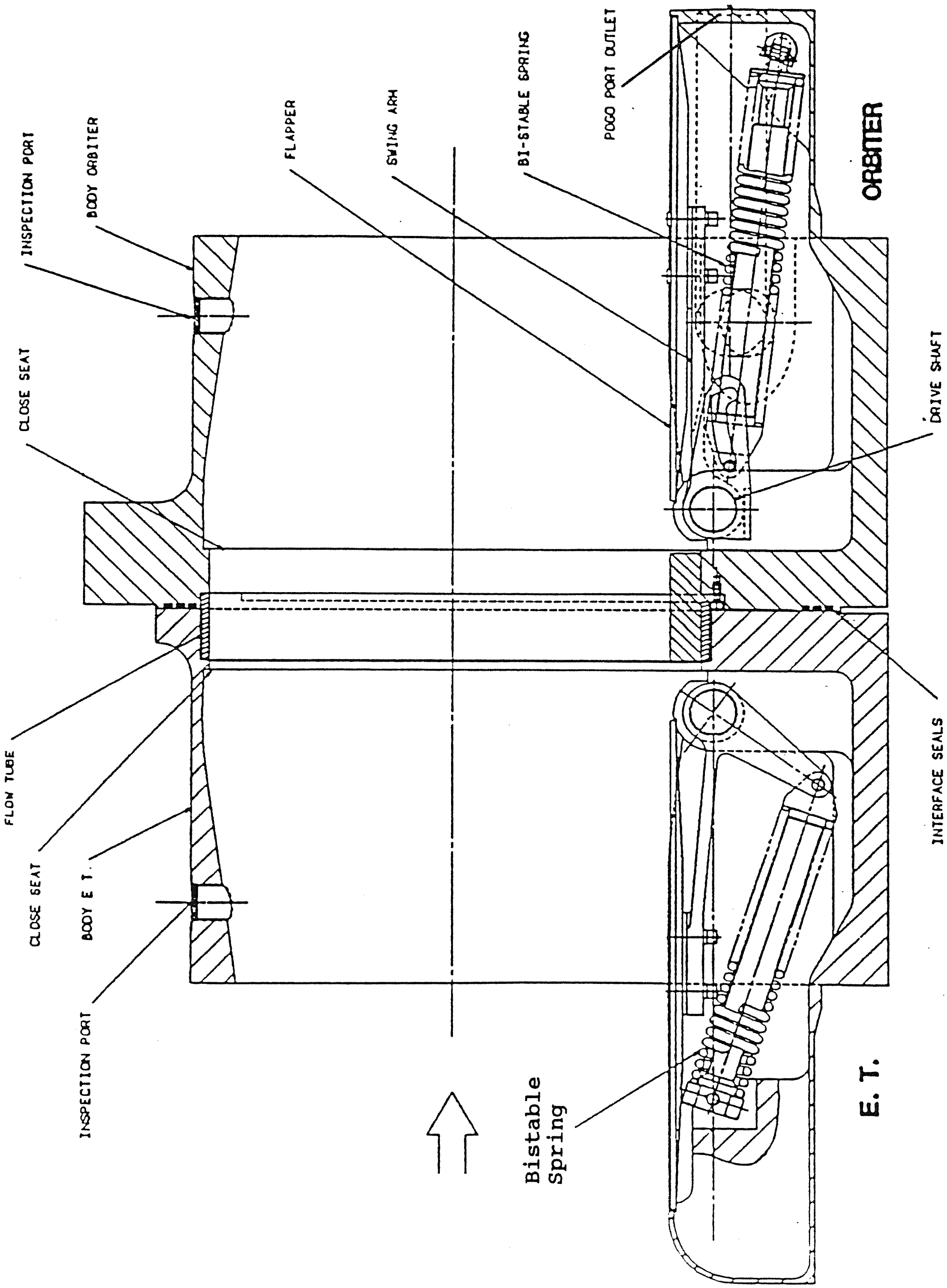
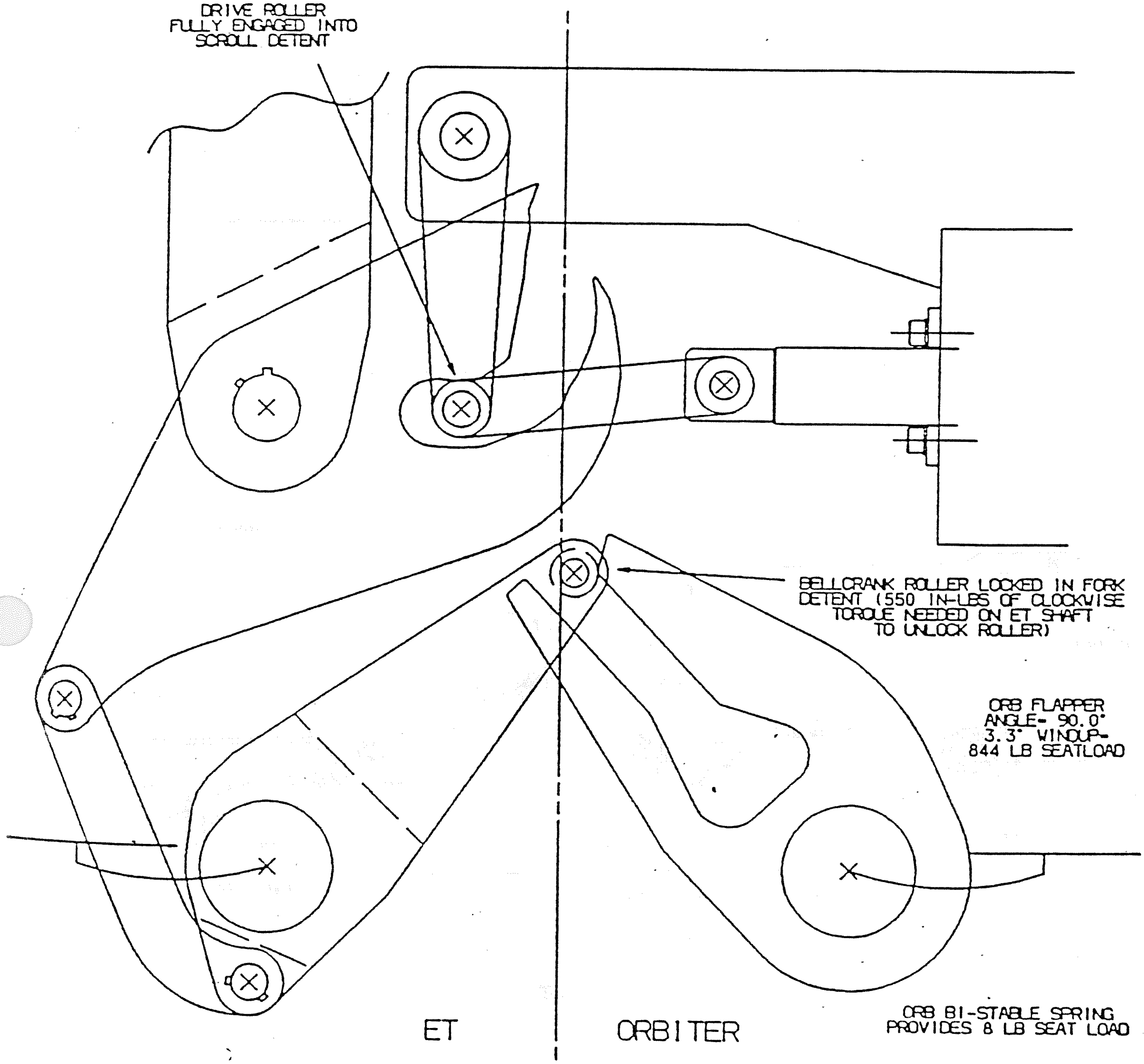

INSPECTION PORT
BODY ORBITER
CLOSE SEAT
FLAPPER
SWING ARM
BI-STABLE SPRING
POGO PORT OUTLET
ORBITER
DRIVE SHAFT
INTERFACE SEALS
FLOW TUBE
CLOSE SEAT
BODY E T.
INSPECTION PORT
Bistable Spring
E. T.

Figure 21



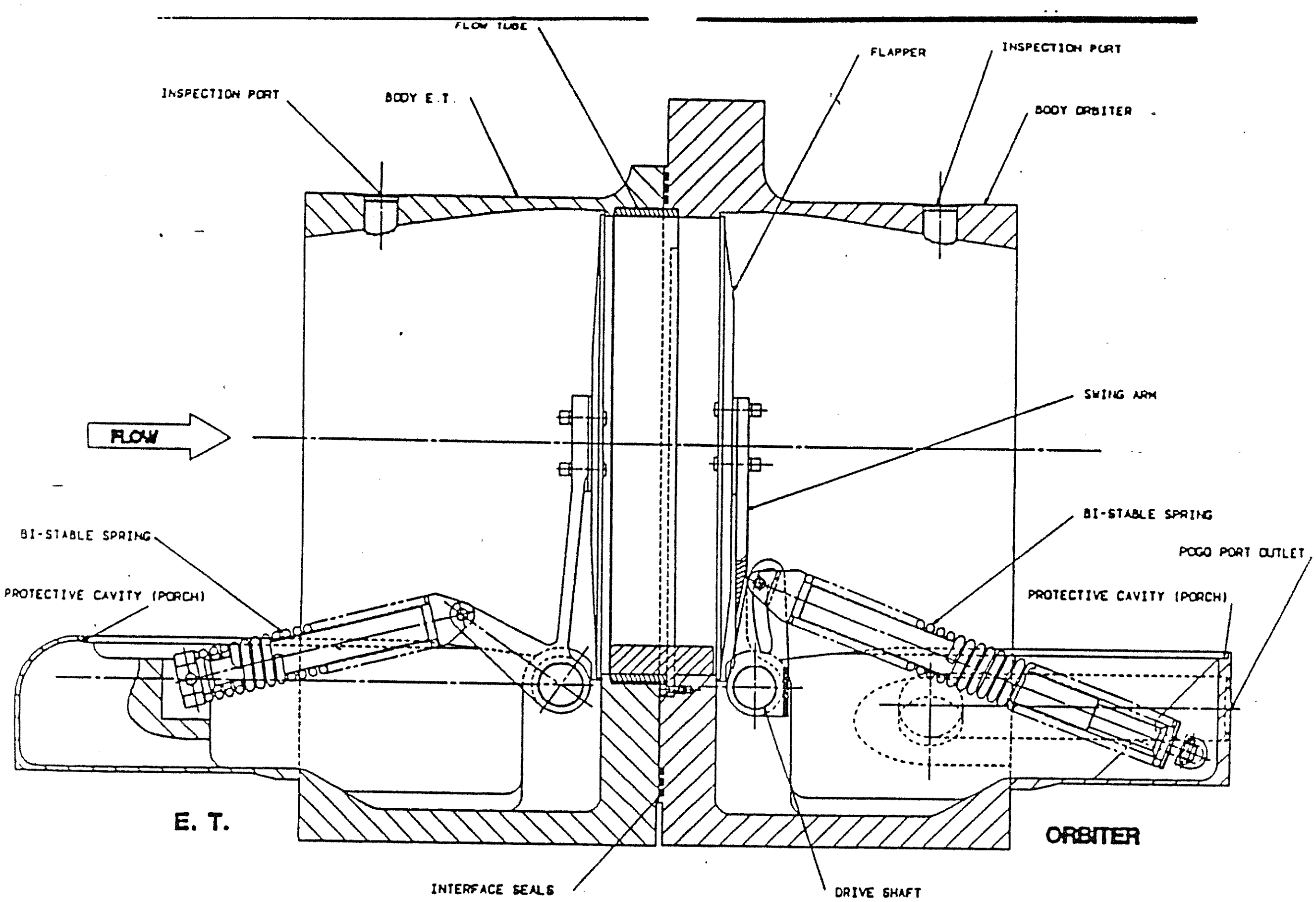
14" DISCONNECT DRIVETRAIN SCHEMATIC

FULL OPEN AND LOCKED FLAPPER POSITION

PISTON EXTENDED 1.800"

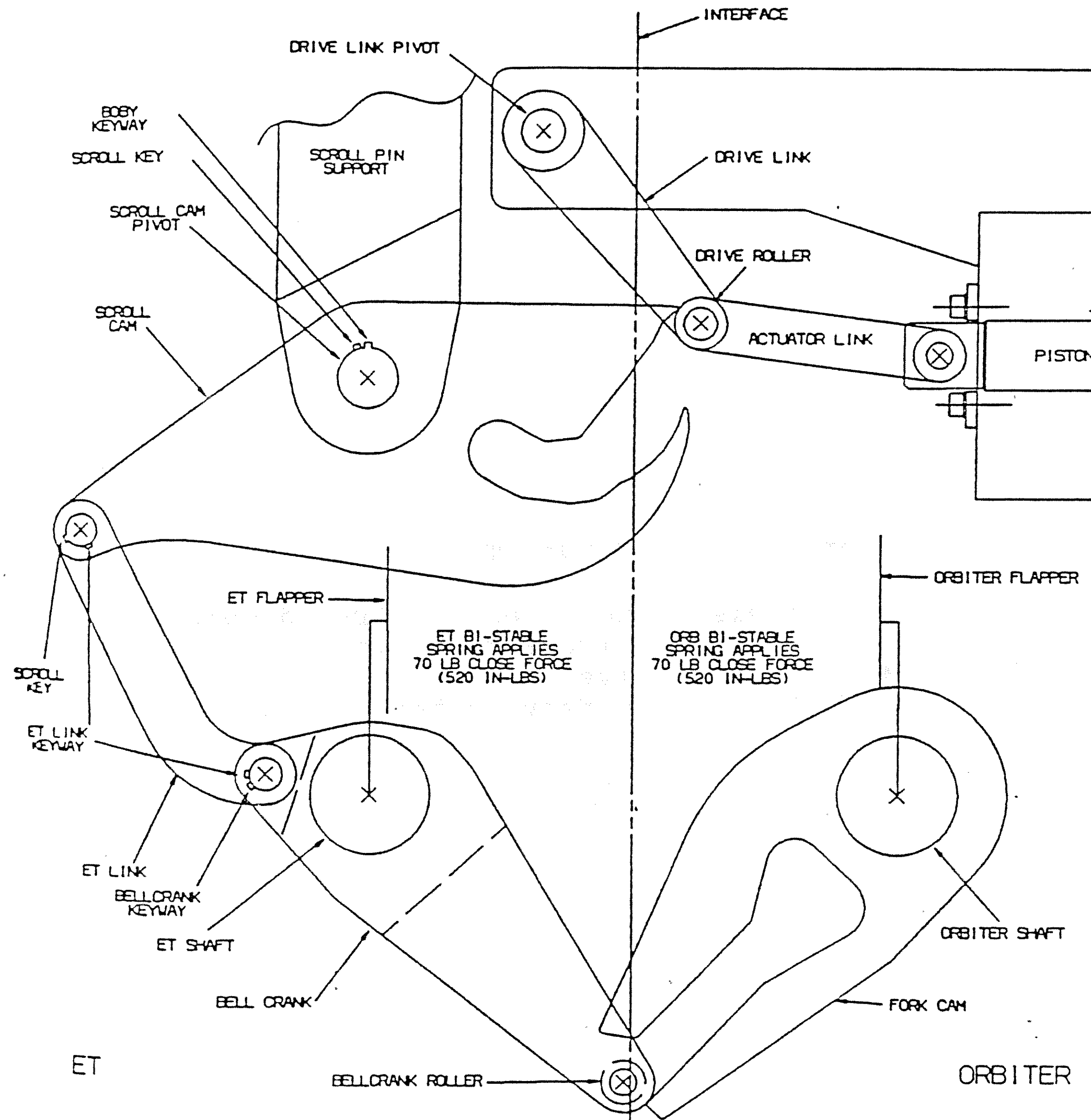
OPEN 13

Figure 22



14 IN. DISCONNECT - CLOSED POSITION

Figure 23



14" DISCONNECT DRIVE TRAIN SCHEMATIC

FLAPPERS IN FULL CLOSED POSITION (0°)

PISTON FULLY RETRACTED

OPEN 1

Figure 24

SECTION 4: EXTERNAL TANK

4.1 Tavis Ullage Pressure Transducers
4.2 ET Vent Valve Stroke Limiter
4.3 Tumble Valve
4.4 ET Range Safety System Removal

4.1 Tavis Ullage Pressure Transducer

Function

The Liquid oxygen (LO_2) and Liquid hydrogen (LH_2) tanks each have four ullage pressure sensors. For flight, three ullage pressure sensors are used to control the ullage pressure by opening or closing flow control valves. The fourth sensor is a spare that can be switched in prelaunch to replace a failed sensor. The LH_2 sensor has a pressure range of 12 to 52 psia. The LO_2 sensor range is 0 to 30 psig. To maintain the desired ullage pressure, the flow control valves on the pressurization lines from the main engines on both the LO_2 and LH_2 are automatically opened if the tank pressure drops out of the control band. The LH_2 flow control valves can be manually opened by the crew if necessary. There is no manual control for the LO_2 flow control valves.

Background

As part of the return to flight activity after STS-51L under the ET electrical design review, an action was assigned to develop an alternate ET ullage pressure transducer for better reliability. The Gulton transducer that is existing is potentiometric and is susceptible to acoustics and vibrations and have had early histories of failures (dropouts, stiction, and shorted turns). The Tavis variable reluctance transducer is solid-state and contains an integral electronics package. Figure 25 shows the comparison between the Gulton and the Tavis.

Comparisons

The Tavis transducer exhibited excellent operations during the development/qualification program. The Gulton transducer is susceptible to potentiometric type failure modes such as stiction, dropout, and shorted turns whereas the Tavis is a variable reluctance type transducer. The accuracy of the Tavis transducer is +/- 1% across the pressure range of 0-30 psid and +/-1% across the temperature range of -65 to +150^{o}F. The Gulton accuracy is +/- 1% in the control bands of 20-22 psid and 0 to 140^{o}F, otherwise the accuracy is +/- 3% The Tavis transducer has less parts and uses an electrical connector instead of splicing. The testing of the Gulton requires a calibration and a ramp test. The Tavis requires a 2 point calibration. The Gulton transducer reliability decreases with use while the Tavis transducer will not malfunction after power is supplied. The weight of the Gulton is 8 oz compared to 25 oz for the Tavis. The input output voltage is the same for each. The Tavis is not susceptible to vibroacoustics whereas the Gulton is.

Current Status (July 30,1991)

The Tavis pressure transducer will be demonstrated on ET-64, 65, and 66 using four active Tavis transducers on the GO_2 system only. Starting with ET-67, three active GO_2 sensors and a dummy sensor will be used. Since the GO_2 fixed orifice will be replacing the flow control valves, the transducers will not be used for active pressure control in flight but will be used for prepressurization control only. The KSC software will be modified to eliminate switching in the spare GO_2 Tavis ullage pressure transducer.

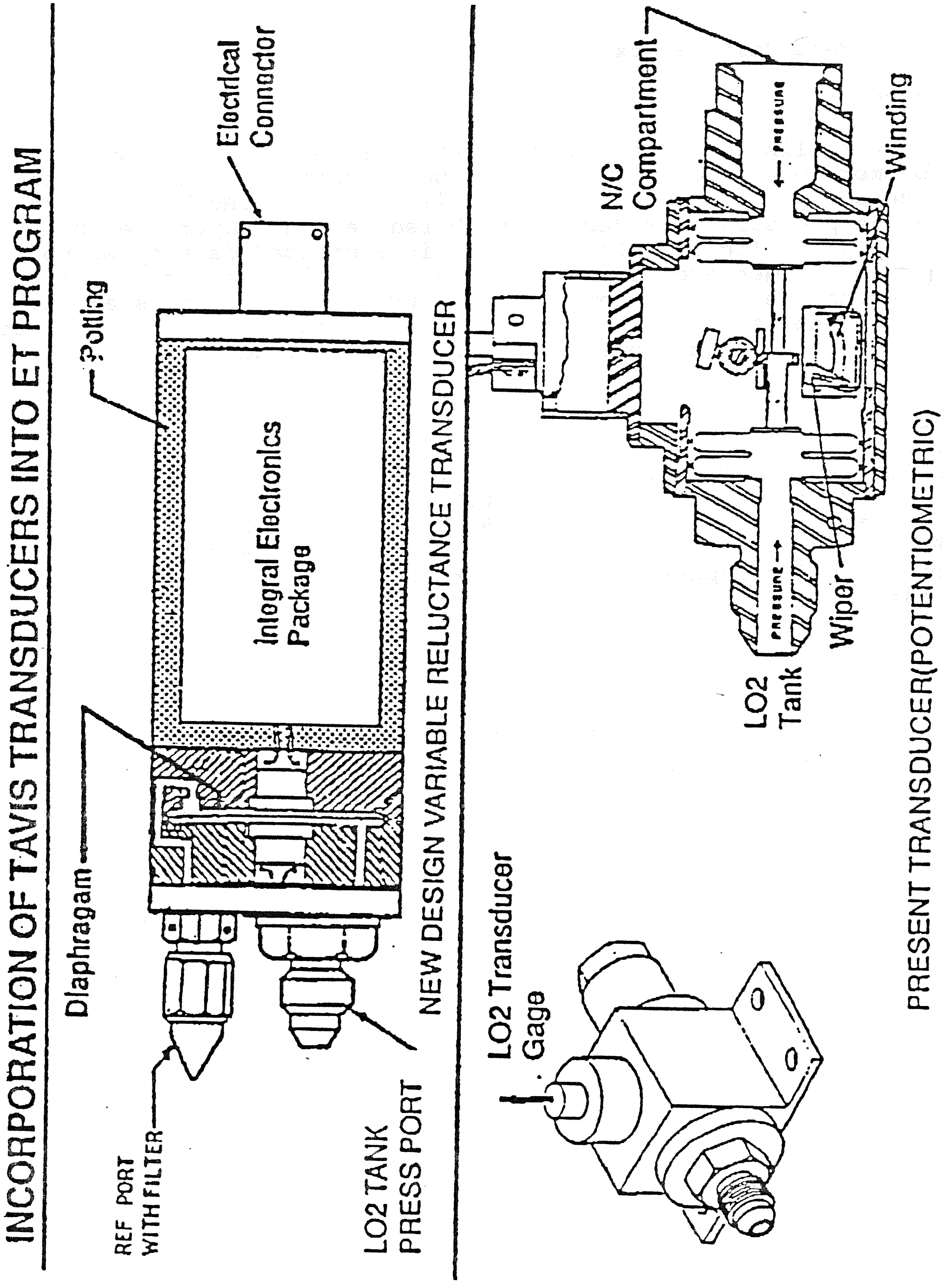

INCORPORATION OF TAVIS TRANSDUCERS INTO ET PROGRAM
Diaphragam
Potting
Electrical Connector
Integral Electronics Package
REF PORT WITH FILTER
LO2 TANK PRESS PORT
NEW DESIGN VARIABLE RELUCTANCE TRANSDUCER
LO2 Transducer Gage
N/C Compartment
Winding
Wiper
LO2 Tank
PRESSURE
PRESENT TRANSDUCER(POTENTIOMETRIC)

Figure 25

4.2 ET Vent Valve Stroke Limiter

Background

The stroke limiter on the ET vent valve is being examined to allow more pounds of oxygen to be loaded during initial tanking on the pad. If more oxygen ullage is vented during LOX loading then the pressure is reduced, which reduces the vapor pressure and cools the liquid. This results in increased density so more pounds of oxygen are loaded per volume. 1.5 is the limit since increasing the stroke above 1.5 the stroke cap may not stay on. The table shows that by increasing the maximum stroke from 1.1 to 1.5, 170 lbs of additional propellant can be loaded.

stroke limit	1.1	1.5	delta
LOX ullage press	0.76	0.27	-0.49
LOX load at SRB ignition	1372751	1374806	+2055
mixture ratio (MR)	6.0227	6.0332	+0.015
fuel bias	968	1023	+55
EPR	4698	4609	-89
propellant for impulse	1588214	1590303	+2089
margin			+170

4.3 External Tank Tumble Valve Removal

Function

The tumble valve is on the external tank (ET) to rock and tumble it after separation to ensure break up in the atmosphere. The 2" tumble valve mounts to the top of the liquid oxygen (LOX) tank. An igniter and pyro booster is used to generate the explosive charge which shears a gate within the valve. This then allows oxygen gas to flow through the valve and exit the ET through an external nozzle. The purpose of the valve is to tumble the ET after separation to reduce the ET impact footprint. Figures 26 and 27 show the tumble system.

Background

STS-30 was the first flight that the tumble valve was disabled by disconnecting cables in the ET intertank area. This removed a potential Criticality 1 hazard of the valve failing open early in ascent and depressuring the LOX. Monte Carlo studies of ET breakup during entry showed that even without the tumble valve operating, the chance of hardware breakup over a populated area was minimal. The flight techniques agenda evaluated the operations of not using the tumble valve. (Refer to FT agendas post STS-30 to review the evaluation results). During STS-31, the ET breakup without using the tumble valve was evaluated. The breakup was tracked by VHF radars in Hawaii and Maui and an optical system (ARGUS aircraft and AMOS/MOTIF). The results noted that the ET tumbles even if the system is disabled but is less. The measured cloud footprint was less than predicted, and disabling the ET tumble valve is acceptable for 28.5 and 57 degree inclination missions. The procedures to disable include, disconnecting a wire in the intertank and removing the pyro system. The ET tumble valve had been used on all flights up to STS-30. It failed to open on STS-1 for unknown reasons, and it was intentionally disabled on STS-7 for photography DTO opportunity. Use of the valve did not impact the ET footprint, and therefore the danger of an inadvertant depressurization of the LOX tank is greater. The tumble valve will be removed on subsequent ETs.

Current Status (as of July 2, 1991)

For those tanks whose nose cones have been closed out for flight, the tumble valve circuitry will be disabled. For those tanks which have not had their nose cone closed out for flight, the Nasa standard initiator/detonator will also be removed. The ET tumble system may be removed completely if there are no impacts in the ET production cycle. Removing the tumble system will decrease risk to vehicle and crew by eliminating criticality one components including detonator, pressure cartridge, tumble valve, etc. The results of these missions indicated that ET rupture altitudes were between 222K feet and 251K feet. The observed rupture altitudes showed no differences in active/inactive tumble systems (T/S's). Using the maximum flight derived ruptured altitude (3 sigma dispersions and 95% confidences), ET footprints without T/S met the range safety requirements i.e. 200 nm off all foreign land masses and 25 nm off a U.S. land mass. The benefits to eliminate the ET T/S completely include in addition to the elimination of several crit 1 for crew/vehicle, provisions for an additional 53 pounds of payload. The cost savings for the ET tumble system hardware is approximately $43K/tank.

Removal of the T/S is to be implemented in three phases (see Figures 28, 29, and 30) i.e. Phase I: Remove the pyro cartridge/detonator and plug the boss on the tumble valve. Rework the cable harness in the nosecone and remove the connector and wiring to the nearest clamp. Phase II: (Inline build tanks ET-65 through ET-80) Remove the tumble valve, pyro cartridge/detonator, and rain shield. Modify the forward ogive fitting and add a blank off patch for the tumble valve port. Remove the cable harness from the NSI to the intertank disconnect panel. Phase III: (component detail build tanks ET-81 and up) Remove cable harness from the NSI to the orbiter interface. Remove the switch module from the range safety system enclosure. Replace the forward ogive fitting. There will be no tumble valve mounting provisions and the manifold and flex joint will be replaced and a redesigned structural support for the level sensor mast will be used.

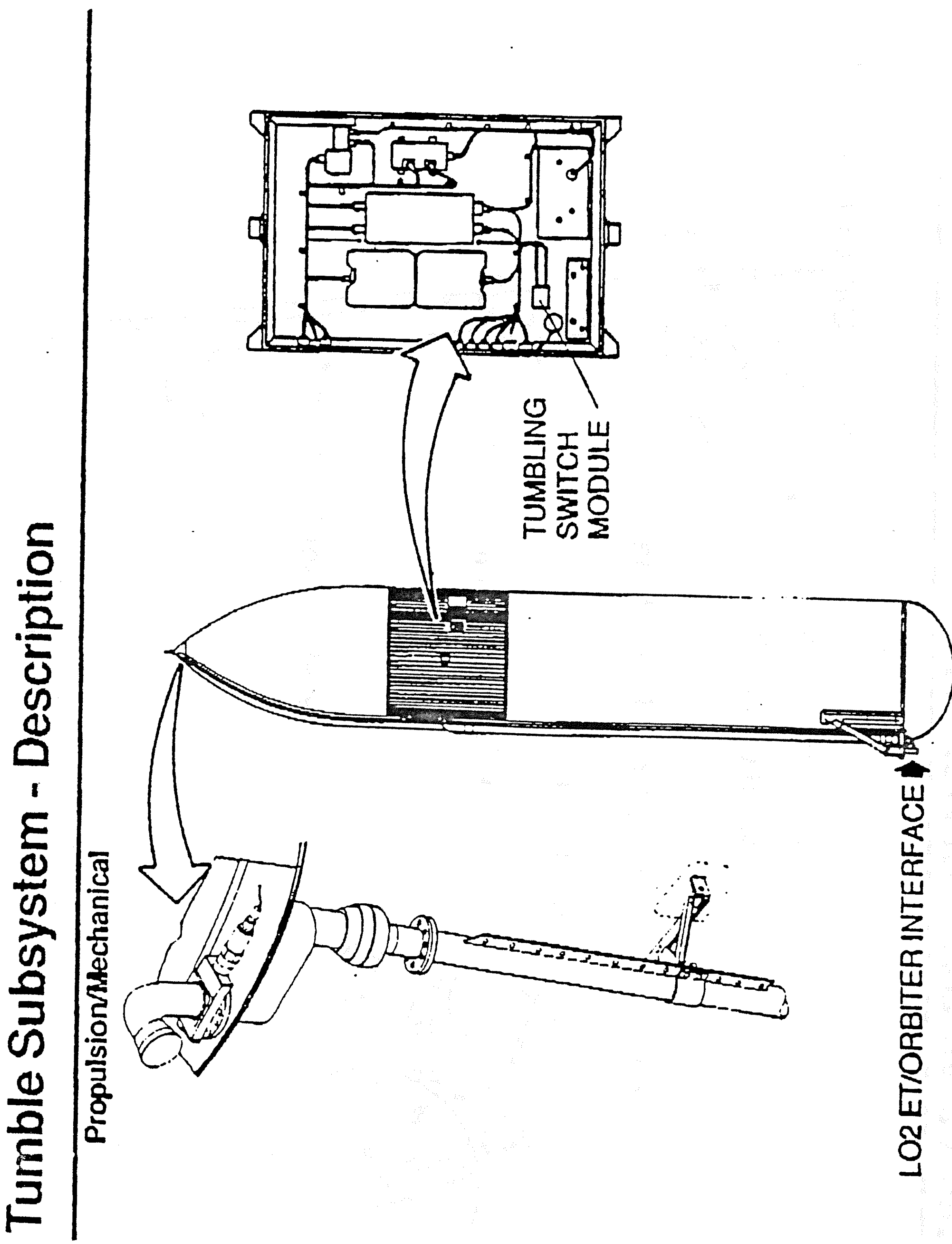

Tumble Subsystem - Description
Propulsion/Mechanical
TUMBLING SWITCH MODULE
LO2 ET/ORBITER INTERFACE

Figure 26

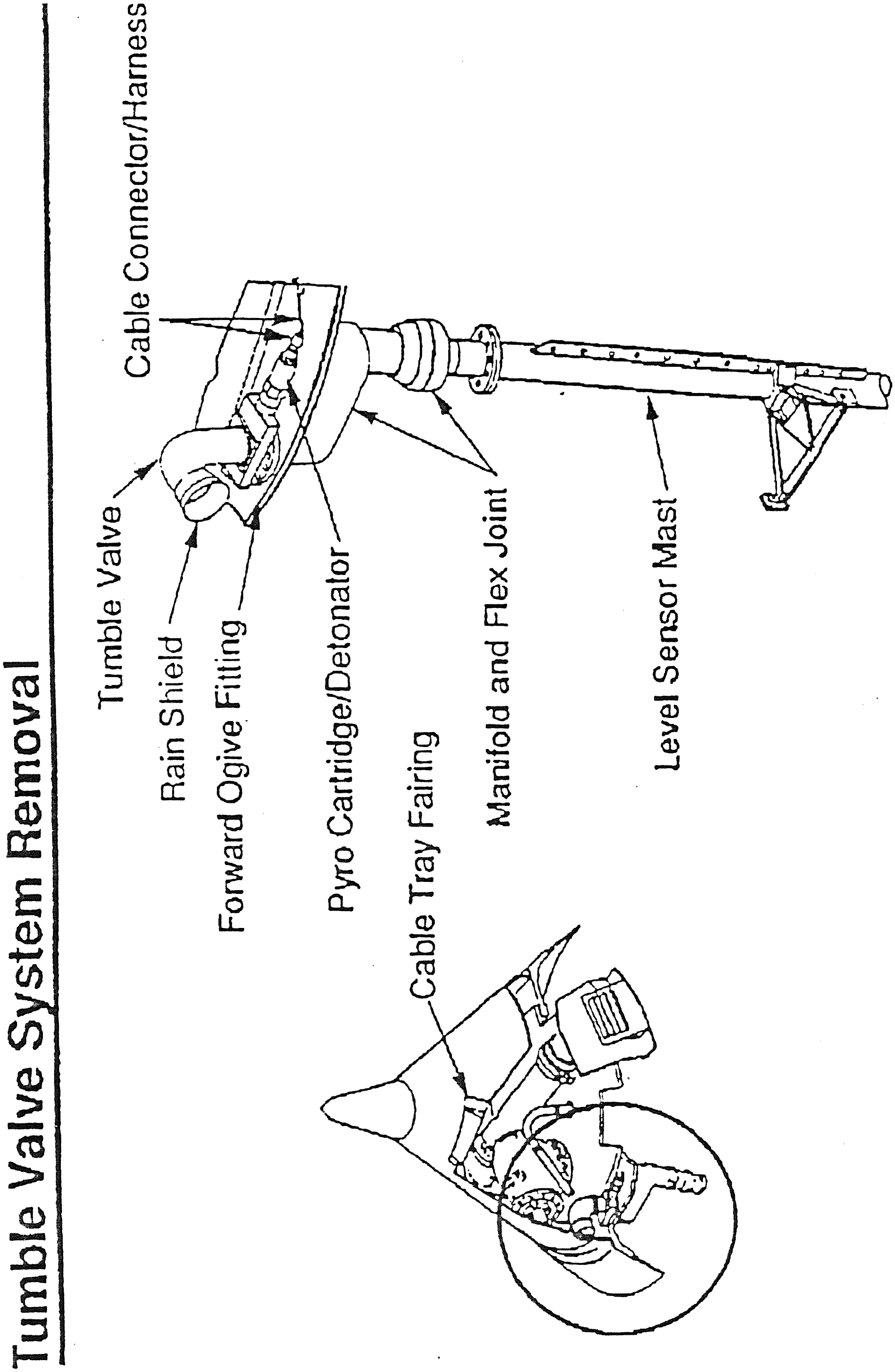

Tumble Valve System Removal
Cable Connector/Harness
Tumble Valve
Rain Shield
Forward Ogive Fitting
Pyro Cartridge/Detonator
Cable Tray Fairing
Manifold and Flex Joint
Level Sensor Mast

Figure 27

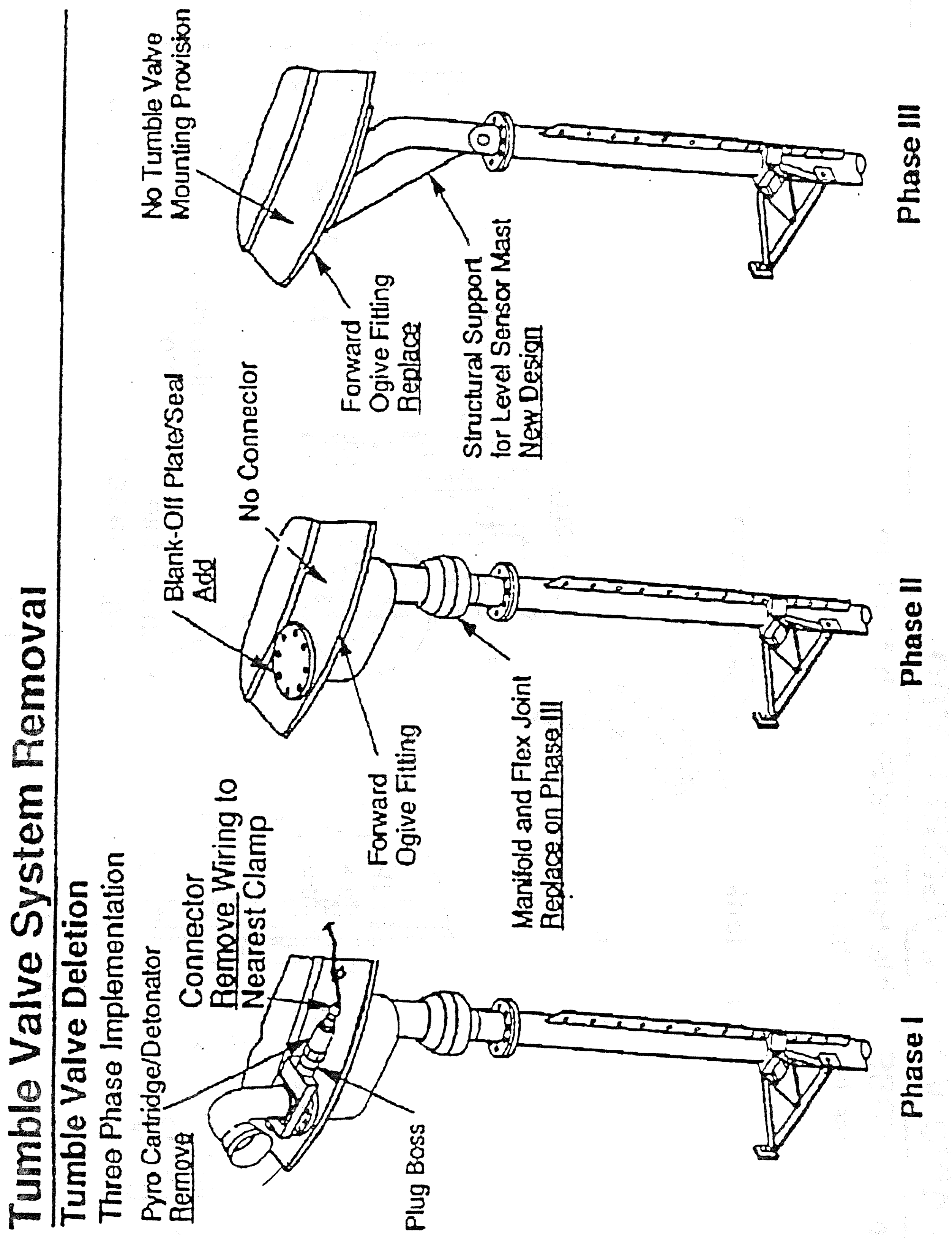

Tumble Valve System Removal
Tumble Valve Deletion
Three Phase Implementation
Pyro Cartridge/Detonator
Remove
Connector
Remove Wiring to
Nearest Clamp
Plug Boss
Phase I
Forward
Ogive Fitting
Manifold and Flex Joint
Replace on Phase III
Blank-Off Plate/Seal
Add
No Connector
Phase II
Forward
Ogive Fitting
Replace
Structural Support
for Level Sensor Mast
New Design
No Tumble Valve
Mounting Provision
Phase III

Figure 28

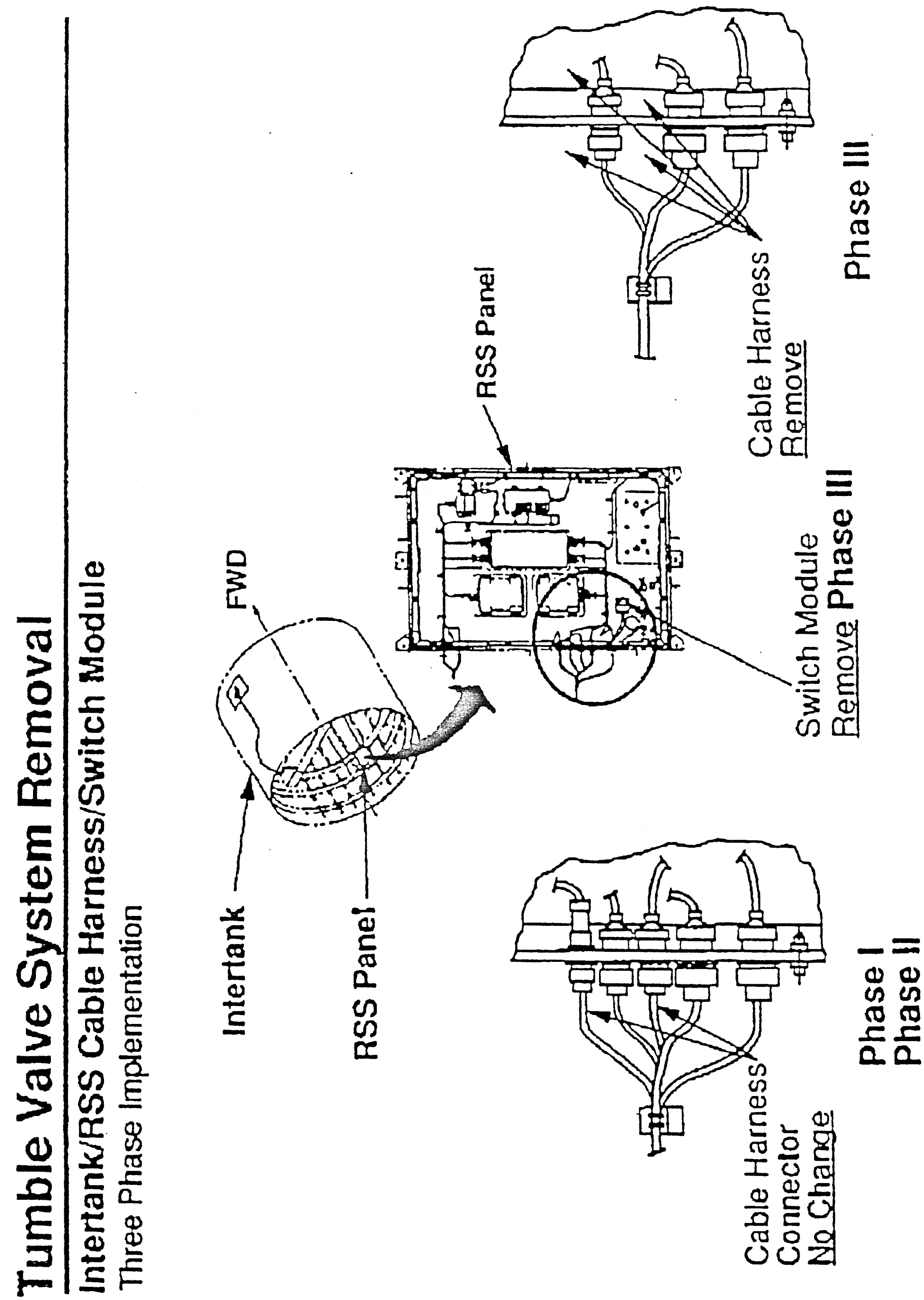
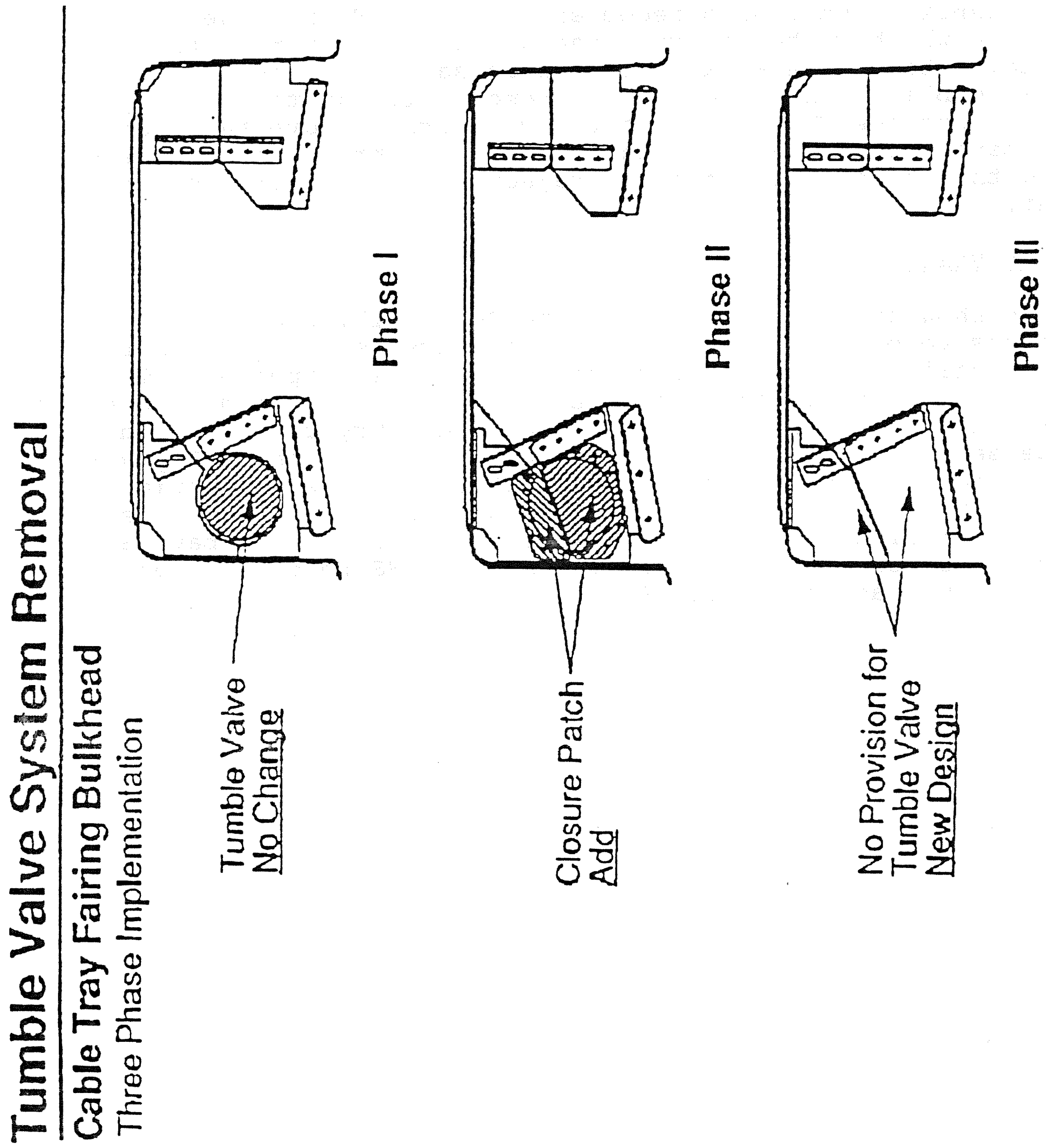

Tumble Valve System Removal
Intertank/RSS Cable Harness/Switch Module
Three Phase Implementation
Intertank
RSS Panel
FWD
RSS Panel
Switch Module
Remove Phase III
Cable Harness
Remove
Phase III
Cable Harness
Connector
No Change
Phase I
Phase II

Figure 29



Tumble Valve System Removal
Cable Tray Fairing Bulkhead
Three Phase Implementation
Tumble Valve
No Change
Phase I
Closure Patch
Add
Phase II
No Provision for
Tumble Valve
New Design
Phase III

Figure 30

4.4 ET Range Safety Removal

Background

A Presidential commission recommended that NASA and the Air Force reexamine whether the destruct package on the external tank might be removed to assure a balance between safety of the crew and the general public. The intent is to remove all ignition and pyro devices from the ET to protect the crew from inadvertant ignition while continuing to ensure that the ET will break up if necessary should the vehicle divert from trajectory during first stage flight.

Current Status

Studies show that destruction of the SRB's will cause case fragments to penetrate the LH_2 tank in first stage, but that no direct infliction of damage to the LOX tank is expected. The ET LH_2 RSS is not required because the LH_2 tank will be destroyed by the SRB RSS in the first stage and by the entry dynamic loading in the second stage. The benefits to eliminate the ET LH_2 RSS include reducing the risk to the orbiter and crew, potential savings at KSC due to reduced launch preparation effort, coss savings for ET hardware, provide additional 38 lbs of payload, reduce a number of crit I items. The LH_2 RSS removal is under further study as is the LO_2 RSS.

SECTION 5: ENHANCED MASTER EVENTS CONTROLLER

5.1 Enhanced Master Events Controller

Function

The Master Events Controller MEC sends commands to the pyrotechnic initiator controller (PIC) for pyro ignition to do such things as separate the SRB's, ET, and to pyrotechnically separate the SRB holddown posts. The controller is the interface unit between the GPC and the orbiter, ET, and SRB pyrotechnic and control devices. The function of the MEC is to receive digital commands from the GPC's and to buffer, compare, and execute those commands through sequential hardware during launch, SRB SEP, and ET SEP. The MEC fires the PICS for SRM ignition, SRB holddown release, T-0 umbilical retraction, and the ET GH_2 vent arm retraction. It also controls SRB staging i.e. after the chamber pressure drops below 50 psi in both SRB's the SRB RSS is powered down and safed, and the recovery system is activated. At the appropriate time, the SRB TVC's are deadfaced, the bolts at the attachment points are severed, and the SRB SEP motors are ignited. At MECO the MEC controls the ET separation sequence. The tumble valve is opened by a single pyro function, the ET umbilicals are unlatched and retracted and the bolts at the attachment points are severed. After ET SEP, the MEC functions are complete and the MEC is powered down.

Comparisons

The EMEC is designed to be compatible with the existing software and hardware interfaces, although it incorporates significant modifications compared to the MEC. It is more reliable and implements improvements in operation, testing, and checkout. The EMEC is physically and functionally interchangeable with the MEC and needs no interface modification to operate in the Space Shuttle environment. The electrical inputs and outputs will be the same. Both use the existing MIA inputs and PIC outputs. The discrete outputs will be electrically identical to the MEC. The mechanical and electrical interfaces are the same. The footprint is not identical as the EMEC uses the same type but fewer orbiter mounting holes. The EMEC will have an additional test connector which will allow additional ATP testing and fault isolation. The wire harness is replaced with a master interconnect board where 1500 wires are eliminated and hand wiring is eliminated. The parts count is reduced from 7328 to 1900. This improves the thermal and vibrational characteristics. The MEC has a two subcore design i.e. Core A/Core B. The EMEC has a two subcore design where each subcore has eight modules and each module has multiple identical subsections. In terms of Critical Command Voting, both the EMEC and MEC will perform functions based on a 2 of 4 vote (provided there is not a 2 on 2 split). The EMEC starts the skew timer on the receipt of second command regardless of channel. The MEC will start the skew timer on channel 1 or 2 for core A, channel 3 or 4 for core B. For Non-critical command voting, the EMEC will perform functions based on a 2 of 4 vote. The MEC will perform functions based on 2 of 3 vote. For the ET tumble valve, the EMEC qualifies the ET tumble valve arm command with fire command. The MEC does not qualify the ET tumble valve arm command. The first four bits of return data word indicate the last channels commanded. The EMEC command channel bits are only reset by the power cycle, master reset, and Operational Bite (OPB) command. The MEC command channel bits are reset by most byte commands. The response will differ between the MEC and EMEC depending on previous command sequence. For the Wrap Test, the EMEC will return command data word (RDW) if the NCR wrap is commanded with no NCR command set. The MEC RDW is not effected by NCR commands set previously. For the PIC capacitor voltage simulation, the EMEC contains 8 PIC's, and the MEC contains ten PIC's (2 are unused, not connected). The EMEC returns all zero pattern (no voltage) for in RDW 5 for unused PIC channels. The MEC returns simulated voltage in RDW 5 for unused PIC channels. For the reset of critical commands the EMEC will set critical commands in both cores based on 2 of 4 channel vote. For the MEC, channels 1/2 will reset core A, channel 3 or 4 to set core B. The CR command set/reset during logic testing with dropped data busses will result in differences betweeen the EMEC and the MEC. For the nominal case, the pre-flight bite (PFB) response will be identical for the MEC and EMEC. The MEC requres firsts pass "cleansing" PFB prior to obtaining accurate data, following power up. The EMEC provides accurate data on the first request. The EMEC will be delivered JSC September 1991 to test in the SAIL. Delivery to the Cape by January 1992 for the first return to flight on 102 STS-50 where a mixed MEC and EMEC will be flown.

APPENDIX B
SYSTEMS BRIEFS ORIGINATORS

SB No.		Page	Originator
	SECTION 1 SSME		
1.1	SSME OVERVIEW	1.1-1	COIL
1.2	SSME PROPELLANT FLOW	1.2-1	STOWE
1.3	SSME GUIDANCE CONTROLLED THROTTLING	1.3-1	JENKINS
1.4	TURBOMACHINERY FUNDAMENTALS	1.4-1	REDING
1.5	SSME MANUAL THROTTLING/ MAXIMUM THROTTLES	1.5-1	STEIN
1.6	SSME VALVE SCHEMATICS	1.6-1	KWIATKOWSKI
1.7	SSME PURGE/ACTUATOR OPERATIONS	1.7-1	MURRAY
1.8	LO_2 PREVALVE CLOSURE AT SSME SHUTDOWN	1.8-1	COIL
1.9	SSME FASCOS SYSTEM (BLOCK 1)	1.9-1	MARKLE
1.10	SSME AUTO SHUTDOWN REDLINES AND REASONABLENESS TESTS	1.10-1	KWIATKOWSKI
1.11	SSME EIU HARDWARE	1.11-1	MARKLE
1.12	SSME CREW CONTROLS	1.12-1	KWIATKOWSKI
1.13	SSME AVIONICS INTERFACES	1.13-1	JENKINS
1.14	SSME CONTROLLER HARDWARE (BLOCK 1)	1.14-1	JENKINS
1.15	SSME "STARTBOX" VIOLATION TURNAROUND REQUIREMENT	1.15-1	SHIFFLETT
1.16	SSME POGO SUPPRESSION SYSTEM	(TBS)	REDING
1.17	SSME SENSOR CONFIGURATIONS	(TBS)	STOWE

SB No.		Page	Contact
	SECTION 2 ET		
2.1	ET OVERVIEW	2.1-1	HAZLE
2.2	ET PROPELLANT LOADING	2.2-1	HAZLE
2.3	ET VALVES SCHEMATICS	2.3-1	HALYARD
2.4	ET LH_2/LO_2 LOW LEVEL CUTOFF BRIEF	2.4-1	MURRAY
2.5	EXTERNAL TANK SEPARATION	2.5-1	STALLARD
	SECTION 3 MPS		
3.1	MPS OVERVIEW	3.1-1	KWIATKOWSKI
3.2	MPS HELIUM SYSTEM DESCRIPTION	3.2-1	GIBB
3.3	MPS VALVES SCHEMATICS	3.3-1	HALYARD
3.4	MPS LOW LO_2 AND LH_2 NPSP	3.4-1	HALYARD
3.5	MPS PROPELLANT DUMP	3.5-1	MURRAY
3.6	MPS VACUUM INERTING	3.6-1	DINGLER
3.7	CREW CONTROLS/DISPLAYS	3.7-1	JENKINS
3.8	MPS ORBITER/ET UMBILICALS OVERVIEW	(TBS)	CALHOUN
	SECTION 4 SRB/SRM		
4.1	SRB OVERVIEW	4.1-1	HAZLE
4.2	SRB CASING	4.2-1	HAZLE
4.3	SRM PROPELLANT	4.3-1	GIBB
4.4	SRM NOZZLE	4.4-1	HAZLE
4.5	SRM IGNITER	4.5-1	HAZLE
4.6	SRB HYDRAULIC POWER UNIT AND TVC SYSTEM	4.6-1	GIBB

SB No.		Page	Contact
	SECTION 5 MEC		
5.1	MASTER EVENTS CONTROLLER OVERVIEW	5.1-1	. . . RIGBY
5.2	MASTER EVENTS CONTROLLER	5.2-1	. . . RIGBY
5.3	ENHANCED MASTER EVENTS CONTROLLER OVERVIEW	5.3-1	. . . . SHIFFLETT
	SECTION 6 VEHICLE OPERATIONS		
6.1	HELIUM SIGNATURE TEST	6.1-1	. . . . DWYER
6.2	SWITCH REDUNDANCY MANAGEMENT . . .	6.2-1	. . . . JENKINS
6.3	I-LOAD ARMING MASS AND ONBOARD VEHICLE MASS CALCULATION	6.3-1	. . . . CALHOUN
6.4	LO_2 GEYSER PREVENTION DURING LOADING	6.4-1	. . . . CALHOUN
	SECTION 7 GENERAL TOPICS		
7.1	LIQUID ROCKET FUNDAMENTALS	7.1-1	. . . . HAZLE
7.2	SOLID ROCKET MOTOR FUNDAMENTALS	7.2-1	. . . . COIL
7.3	STORAGE OF PRESSURIZED GASES AND CRYOGENIC FLUIDS	7.3-1	. . . . SEQUEIRA
7.4	COMMON ORBITER CONNECTORS	7.4-1	. . . . J. FOX
7.5	ELECTRON BEAM WELDING	7.5-1	. . . . HOWARD/ STEIN
7.6	PYROTECHNIC INITIATOR CONTROLLER	7.6-1	. . . . D. RANDALL
7.7	ADVANCED SOLID ROCKET BOOSTER . .	7.7-1	. . . . HAZLE
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	ADVANCED/HISTORICAL PROJECTS . . .	A-1	. . . . . . FULLER
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