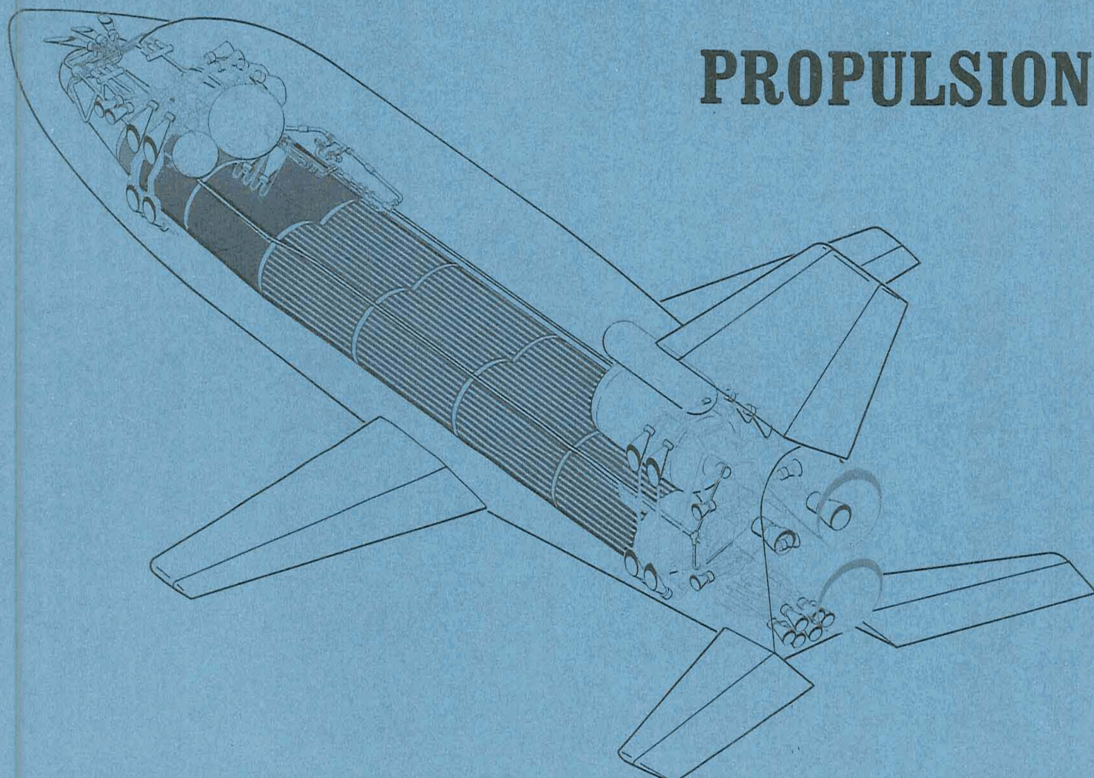
REPORT MDC E0302

NASA CR 115000

SPACE SHUTTLE LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM DEFINITION



SUBTASK B REPORT

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY • EAST

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SPACE SHUTTLE LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM DESIGN DEFINITION

CONTRACT NO. NAS 9-11012

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SUBTASK B REPORT

Prepared by:

W.M. Green

T.C. Patten

Approved by:

L.F. Kohrs, Manager, Propulsion

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY - EAST

Saint Louis, Missouri 63166 (314) 232-0232

MCDONNELL DOUGLAS CORPORATION

ABSTRACT

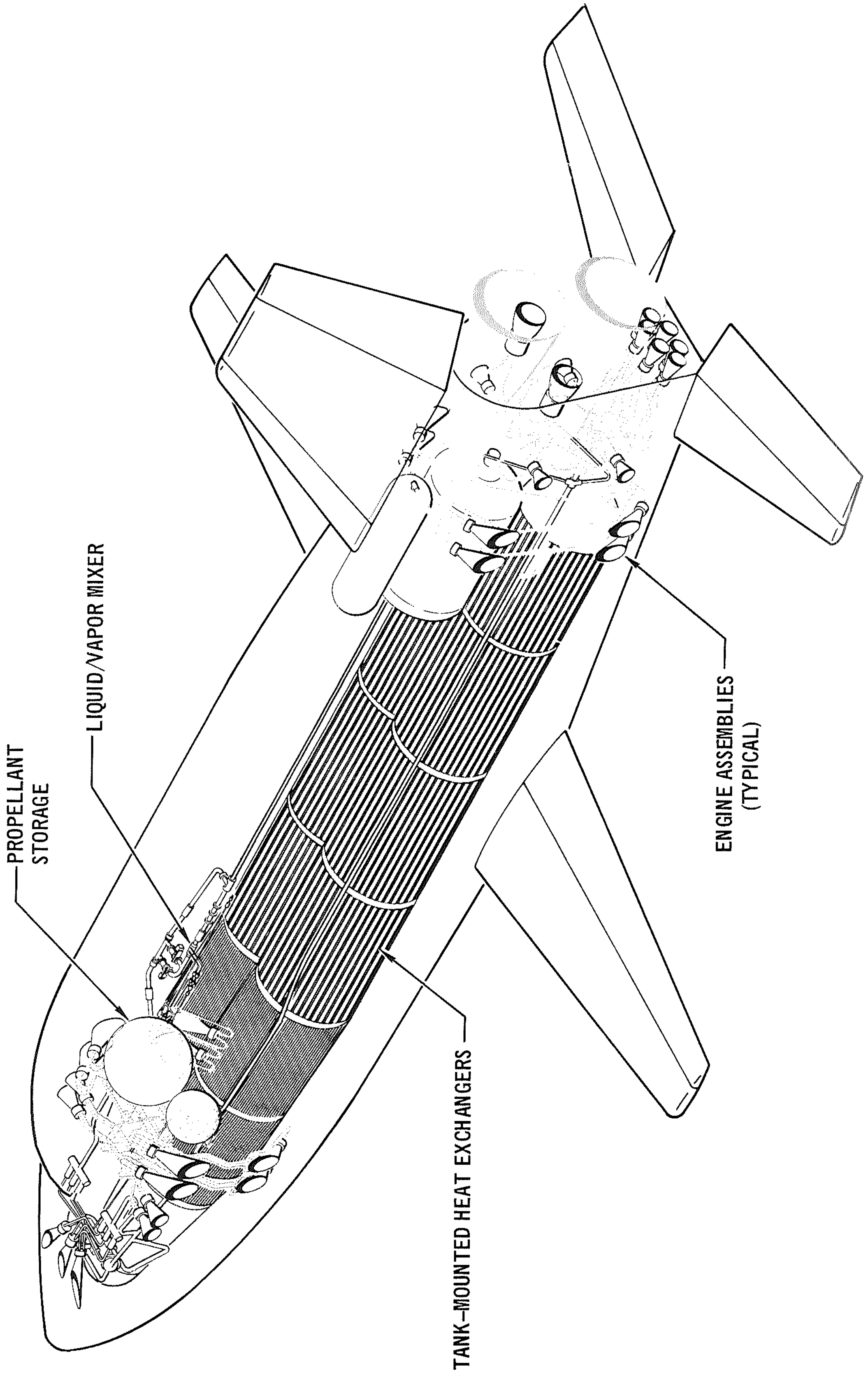
This report describes the preliminary design effort (Subtask B) of a study to define the most attractive low pressure, oxygen/hydrogen auxiliary propulsion system (APS) for NASA space shuttle boosters and orbiters. The study was performed for the National Aeronautics and Space Administration, Manned Spacecraft Center (MSC), Houston, Texas, under Contract No. NAS 9-11012.

The study program was divided into two phases. The first, Subtask A, was a conceptual subsystem definition phase to identify APS concepts best suited to each of two baseline shuttle booster and orbiter vehicles. The Subtask A results are summarized in Report MDC E0303. The second phase, Subtask B, (described in this report) was a preliminary design of the selected subsystems to establish greater understanding of subsystem design and operation. A summary of the overall study effort is provided in Report MDC E0293 and a design handbook containing detailed descriptions of the selected booster and orbiter APS concepts is provided in Report MDC E0301.

The selected orbiter APS concept incorporates auxiliary liquid propellant storage, passive tank mounted heat exchangers and a constant density control scheme. The heat exchangers condition the main engine tank resupply propellant and have been mounted directly to the longitudinal stiffeners on the tank wall. The stiffness associated with the heat exchanger tubes allowed a reduction in tank rib height (O_2 tank only) and consequently reduced the heat exchanger weight penalty. Propellant resupply requirements were minimized by a) using a compartmented O_2 tank to hold main engine tank residual O_2 liquid and b) by operating the system in a mixed mode where liquid-vapor mixing and constant density control are provided only during high usage periods. During low usage, propellant vapors are extracted from the main engine tanks and supplied directly to the engines. This reduces the control complexity required, and although engine inlet conditions vary with tank pressure and temperature fluctuations, the effects on engine performance are small.

The booster APS consists of propellant distribution and engine assemblies. It operates entirely from main engine tank residual propellants, requiring no additional propellant tankage, conditioning equipment or mixing assemblies. Extraction of liquid residual propellants is prevented by g sensitive valves which allow only vapors to be withdrawn.

The APS design as defined during this study provides high performance and high reliability and satisfies all of the design objectives. Technology requirements were identified and relate primarily to component size and dynamic response, factors which generally can be resolved through normal development. The study has shown that a low pressure APS, which is simple in design and operational approach, can fulfill shuttle requirements.



LOW PRESSURE ORBITER APS INSTALLATION

CONTENTS

EFFECTIVE PAGES

1. INTRODUCTION

Auxiliary propulsion will be required for space shuttle attitude and translational control. Operating on the same types of propellant (i.e., oxygen and hydrogen) as the shuttle main propulsion, these subsystems will have a minimum service life of 100 mission cycles without major overhaul or refurbishment. Two basic design approaches have been conceived for the auxiliary propulsion subsystem (APS): a high pressure concept, using turbopumps or turbocompressors to achieve high operating pressure levels, and a low pressure concept, using the main engine propellant tanks as an integral part of the subsystem and operating at main engine tank ullage pressures. This report deals only with low pressure APS concepts. It describes the scope and results of the second phase of a seven month McDonnell Douglas Astronautics Company-East (MDAC-East) study effort under MSC Contract No. NAS 9-11012, titled "Space Shuttle Low Pressure Auxiliary Propulsion Subsystem Definition". The study was performed for the National Aeronautics and Space Administration, Manned Spacecraft Center (MSC), Houston, Texas, under the technical direction of Mr. N. Chaffee. A subcontract was extended to Aerojet Liquid Rocket Company to provide component designs. The study objective was "to conduct preliminary auxiliary propulsion subsystem studies, which (would) generate information and data, for use in the overall shuttle vehicle effort", and which would, "identify attractive APS concepts, define their range of applicability and limitations and identify critical technology areas and development priorities".

The high pressure APS study was conducted by MDAC-East under MSFC Contract No. NAS 8-26248. Technical direction for this effort was provided by the NASA Marshall Space Flight Center (MSFC) at Huntsville, Alabama.

The low pressure APS study program was divided into two phases. The first, Subtask A, was a conceptual subsystem definition phase to identify APS concepts best suited to each of two baseline shuttle booster and orbiter vehicles. The Subtask A results are summarized in Report MDC E0303. The second phase, Subtask B, (described in this report) was a preliminary design of the selected subsystems to establish greater understanding of subsystem design and operation. A summary of the overall study effort is provided in Report MDC E0293 and a design handbook containing detailed descriptions of the selected booster and orbiter APS concepts is provided in Report MDC E0301.

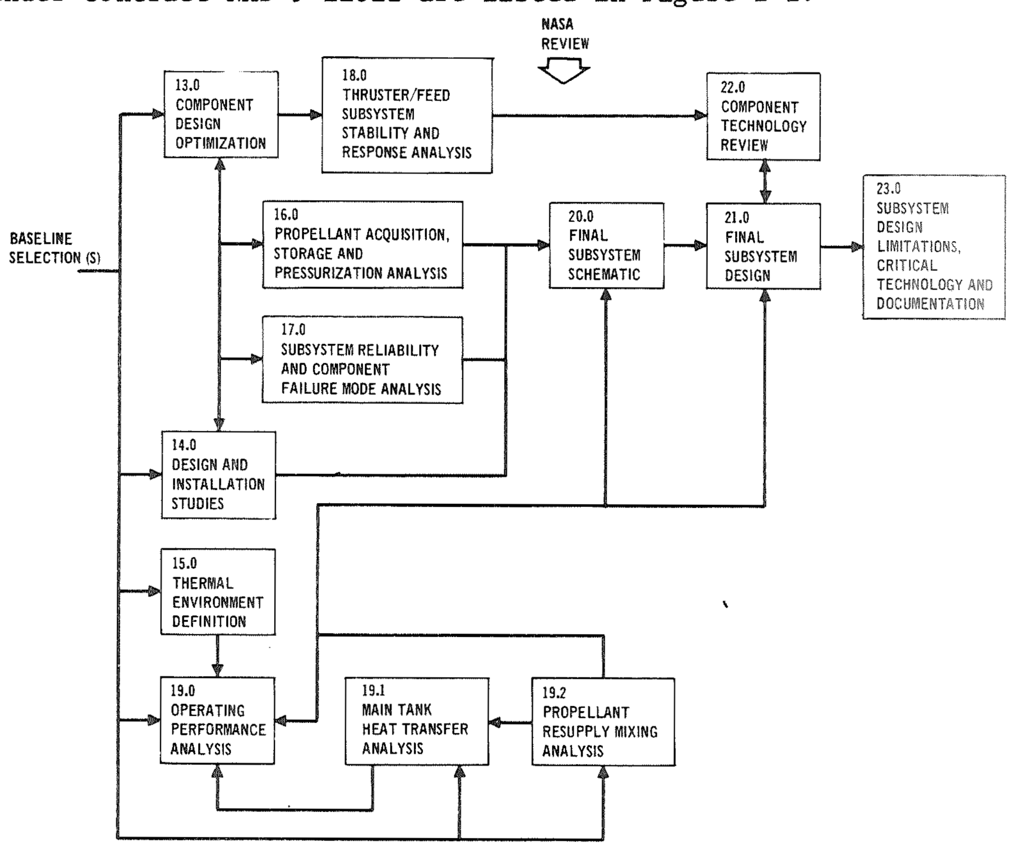
The selected orbiter APS concept requires separate liquid propellant storage tanks to supplement main engine tank residuals. Propellant from the storage tanks is circulated through tubular, passive heat exchangers where it is superheated and injected into the main engine tanks. During major APS burns, warm propellant vapors from the main engine tanks are mixed with additional liquid propellants in a downstream liquid/vapor mixer and supplied to the engines at constant temperature and pressure (constant density). During low usage, propellant vapors are extracted from the main engine tanks and supplied directly to the engines.

The selected booster APS consists of propellant distribution and engine assemblies. These operate entirely from main engine tank residual propellants, requiring no additional propellant tankage, conditioning equipment, or mixing assemblies.

2. STUDY APPROACH

The APS study was divided into two levels of design detail. The first, Subtask A, was broad in scope, and considered many concepts. During that phase, attractive concepts were synthesized, their range of applicability defined, and critical technology areas identified. A description of the Subtask A study and the resulting APS concept selections are presented in a separate report (MDAC-East Report No. MDC E0303).

The second phase, Subtask B, is described in this report. This phase involved preliminary design of the selected APS concepts. Study requirements were updated to reflect revisions in shuttle requirements; APS thrust levels and engine arrangements were defined, and a baseline subsystem design was established. Detailed subsystem performance and component design studies were then initiated. Component design, performance, and transient characteristics were evaluated in detail and these characteristics, along with component technology requirements, were used to update the baseline. A study flow diagram of the Subtask B study effort is shown in Figure 2-1. Detailed study efforts included in the tasks of Figure 2-1 and required under Contract NAS 9-11012 are listed in Figure 2-2.



SUBTASK B PRELIMINARY APS
DESIGN TASK DESCRIPTION FLOW CHART

FIGURE 2-1

1. Definition of instrumentation and controls (amount, type, duty cycles, response, operating ranges).
2. Definition of subsystem, assembly and component operating and nonoperating environment for the various mission phases.
3. Determination of preliminary subsystem, assembly and component weights and volumes.
4. Definition of component life requirements.
5. Determination of propellant conditioning requirements.
6. Determination of thermal control requirements.
7. Identification of new technology required.
8. Determination of subsystem reliability.
9. Determination of a preliminary failure mode and effects analysis.
10. Determination of inherent design and performance limits.
11. Definition of APS/vehicle interfaces.
12. Definition of maintainability requirements.

13. Propellant Storage

    a. Determination of means of storing APS propellants.
    b. Definition of methods of propellant positioning (if necessary).
    c. Determination of the method of loading propellants, e.g., external (from ground) or internal (from main tanks) including use of consumables from the payload compartment (orbiter only).
    d. Determination of the method of propellant egress from APS storage.
    e. Determination of method of pressurizing propellants.

14. Propellant Conditioning

    a. Definition of power requirements.
    b. Determination of required operating duty cycle.
    c. Definition of method of waste gas disposal(if applicable).
    d. Definition of stable operating regimes.

15. Propellant Distribution

    a. Determination of thermal control requirements.
    b. Definition of stability and dynamics characteristics.

16. Thruster

    a. Determination of required duty cycle.
    b. Determination of propellant inlet conditions.
    c. Determination of envelope constraints.

SUBTASK B REQUIRED STUDIES

FIGURE 2-2

3. VEHICLE AND SUBSYSTEM REQUIREMENTS AND CONSTRAINTS

Subtask B attitude control and maneuvering capability requirements for both booster and orbiter elements of the low cross range space shuttle were defined in the revised "Space Shuttle Vehicle Description and Requirements Document (SSVDRD)" dated 1 October 1970. This document is summarized in Appendix F herein. The revised (SSVDRD) vehicle acceleration requirements are tabulated in Figure 3-1 for convenience. SSVDRD failure criteria required that nominal acceleration levels be achieved with one engine out, and minimum (safe) acceleration levels with two engines out.

VEHICLE ACCELERATIONS		MINIMUM	NOMINAL MIN – MAX	MAXIMUM
(BOOSTER)	TRANSLATION, FT/SEC^2	0.0	0.0 – 0.0	0.0
	PITCH, DEG/SEC^2	0.3	0.5 – 1.0	2.0
	YAW, DEG/SEC^2	0.3	1.0 – 1.75	2.0
	ROLL, DEG/SEC^2	0.3	1.0 – 1.75	2.0
(ORBITER)	TRANSLATION, FT/SEC^2 (+X)	0.5	0.65 – 3.0	7.0
	(OTHER)	0.1	0.2 – 0.5	7.0
	PITCH, DEG/SEC^2	0.5	1.0 – 2.0	4.0
	YAW, DEG/SEC^2	0.9	1.3 – 1.7	4.0
	ROLL, DEG/SEC^2	0.5	1.0 – 2.0	4.0
	REENTRY BANK ANGLE (Y–R), DEG/SEC^2	–	1.5*	–
FAILURE CRITERIA – NOMINAL ACCELERATION WITH ONE (1) ENGINE-OUT – MINIMUM (SAFE) ACCELERATION WITH TWO (2) ENGINES-OUT				

*ALL ENGINES OPERATING

VEHICLE ACCELERATION REQUIREMENTS

FIGURE 3-1

Subtask B requirements were similar to those of Subtask A, but differed in several significant areas. The NASA determined that the Subtask B preliminary design effort should be directed only toward the low cross range orbiter and booster vehicles. In addition, the Subtask A approach of defining low pressure APS performance for a number of discrete velocity increments was revised in favor of determining the most favorable distribution of +X axis maneuvers between APS and OMS. Engine failure criteria and mission characteristics were also updated, using current space shuttle study results. Engine locations were modified so that nominal requirements were met with one engine out, in addition to meeting safe requirements with 2 engines out. A requirement for a coordinated yaw-roll bank angle on the orbiter was also established. Orbiter missions were defined as a short, third orbit space station rendezvous and a longer, seventeenth orbit space station rendezvous.

A comparison of the most significant changes in both orbiter and booster vehicle characteristics is shown in Figure 3-2. Main tank operating pressures were reduced and vapor temperatures were increased, resulting in less available propellant vapor residuals. In addition, booster impulse requirements were nearly doubled, and hydrogen liquid residuals were significantly increased. These changes affected subsystem sizing and performance, in terms of reduced available pressure budget and increased tank pressure collapse sensitivity.

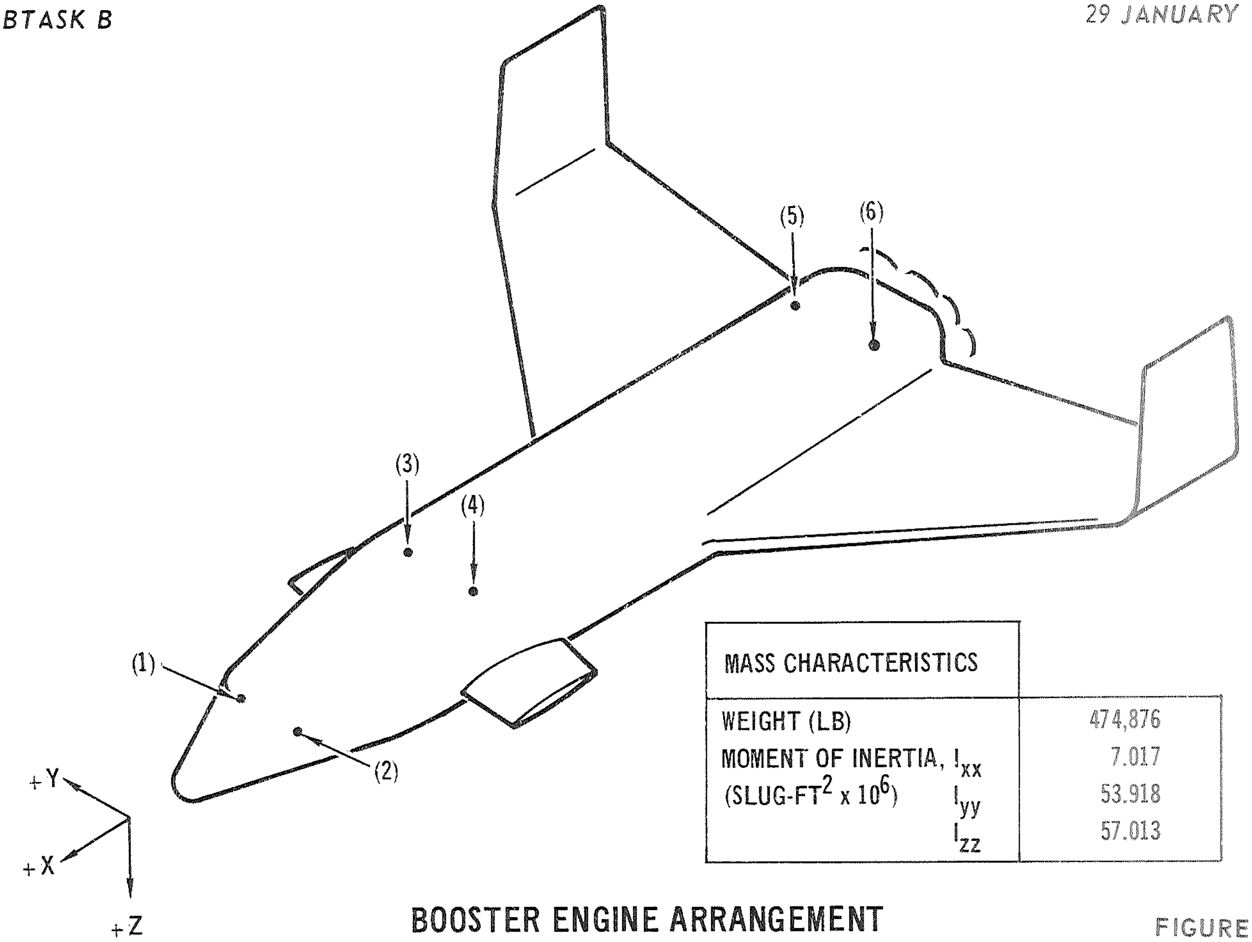
	BOOSTER		ORBITER	
	SUBTASK A	SUBTASK B	SUBTASK A	SUBTASK B
VEHICLE BURNOUT WEIGHT – LB	458,600	474,876	224,900	271,465
MAIN TANK PRESSURE AT BURNOUT – LBF/IN^2A				
H_2	45	26	45	26
O_2	35	26	35	26
MAIN TANK VAPOR TEMP AT BURNOUT – ^oR				
H_2	200	450	200	454
O_2	300	520	300	379
RESIDUAL LIQUID – LB				
H_2	1,816	5,509	511	736
O_2	12,241	11,278	2,392	2,440
MISSION TIME	6 MIN	6 MIN	5 – 7 DAYS	1 – 3 DAYS
TOTAL IMPULSE – LB-SEC	$0.487 (10^6)$	$0.864 (10^6)$	$13.938 (10^6)$	$12.9 (10^6)$

COMPARISON OF SUBTASK A AND B DESIGN DATA

FIGURE 3-2

Studies were conducted to define APS requirements in terms of thrust level, total impulse, and number of engines required. Results of these studies are discussed below. Sensitivity to requirements and design criteria, such as bank angle and engine failure criteria, are discussed in Section 7.

3.1 Booster - All low cross range booster mission requirements were met with an engine arrangement using 2500 lb thrust engines. The arrangement consists of eight yaw engines and twelve pitch-roll engines, shown schematically in Figure 3-3. This arrangement differs from the engine locations specified in the SSVDRD, and was selected because it was more adaptable to a low pressure APS and significantly reduced subsystem weight. Figure 3-4 shows a comparison of booster subsystem design points for both the revised arrangement and for the arrangement specified



MASS CHARACTERISTICS	
WEIGHT (LB)	474,876
MOMENT OF INERTIA, I_{xx}	7.017
(SLUG-FT^2 x 10^6) I_{yy}	53.918
I_{zz}	57.013

BOOSTER ENGINE ARRANGEMENT

FIGURE 3-3

CONCEPT	SSVDRD ENGINE LOCATION	REVISED ENGINE LOCATIONS
SUBSYSTEM OPTIMUMS		
P_C	10.5	11.1
MR	4.0	4.0
ϵ	3.0	2.0
MAX LINE DIAMETER, IN		
H_2	9.3	9.4
O_2	9.7	9.9
SUBSYSTEM WEIGHTS, LB		
PROPELLANT	NO APS WEIGHT PENALTY (OPERATES ON BOOST RESIDUALS)	
ISOLATION VALVES, REG	1392	1151
LINES	1561	608
ENGINES	3072	2404
(TOTAL)	6025	4163

WEIGHT COMPARISON OF ALTERNATE BOOSTER ENGINE LOCATIONS

FIGURE 3-4

in the SSVDRD. Figure 3-4 also provides a comparison of subsystem weights showing that the revised locations have an 1860 lb weight advantage. Engine locations, their function, and installation data are tabulated in Figure 3-5 for each engine group shown in Figure 3-3. A comparison of attitude control acceleration requirements and subsystem performance capabilities is made in Figure 3-6. As this figure shows, selected booster APS performance capabilities exceed all attitude control requirements including all failure mode conditions. The booster impulse require-

LOCATION	MANEUVER	NUMBER OF ENGINES	ENGINE COORDINATES*			DIRECTION COSINE		
			X	Y	Z	X	Y	Z
1	− YAW	4	−425	+93.5	−13	0	−1	0
2	+ YAW	4	−425	−93.5	−13	0	+1	0
3	− PITCH, + ROLL	3	−1120	+151	−149	0	−0.309	+0.95
4	− PITCH, − ROLL	3	−1120	−151	−149	0	+0.309	+0.95
5	+ PITCH, + ROLL	3	−2569	+151	−149	0	0	+1
6	+ PITCH, − ROLL	3	−2569	−151	−149	0	0	+1

*SEE APPENDIX F FOR COORDINATE SYSTEM. ALL DIMENSIONS ARE IN INCHES.

BOOSTER ENGINE LOCATIONS
Twenty 2500 Lb Thrust Engines

FIGURE 3-5

AXIS	CONDITION	ACCELERATION (DEG/SEC^2)	
		REQUIRED	MIN AVAILABLE
+ PITCH	ALL ENGINES OPERATING	−	0.744
	ONE ENGINE OUT	0.50	0.620
	TWO ENGINES OUT	0.30	0.495
− PITCH	ALL ENGINES OPERATING	−	1.120
	ONE ENGINE OUT	0.50	0.935
	TWO ENGINES OUT	0.30	0.747
± YAW	ALL ENGINES OPERATING	−	1.321
	ONE ENGINE OUT	1.00	1.00*
	TWO ENGINES OUT	0.30	0.661
± ROLL	ALL ENGINES OPERATING	−	1.285
	ONE ENGINE OUT	1.00	1.028
	TWO ENGINES OUT	0.30	0.771

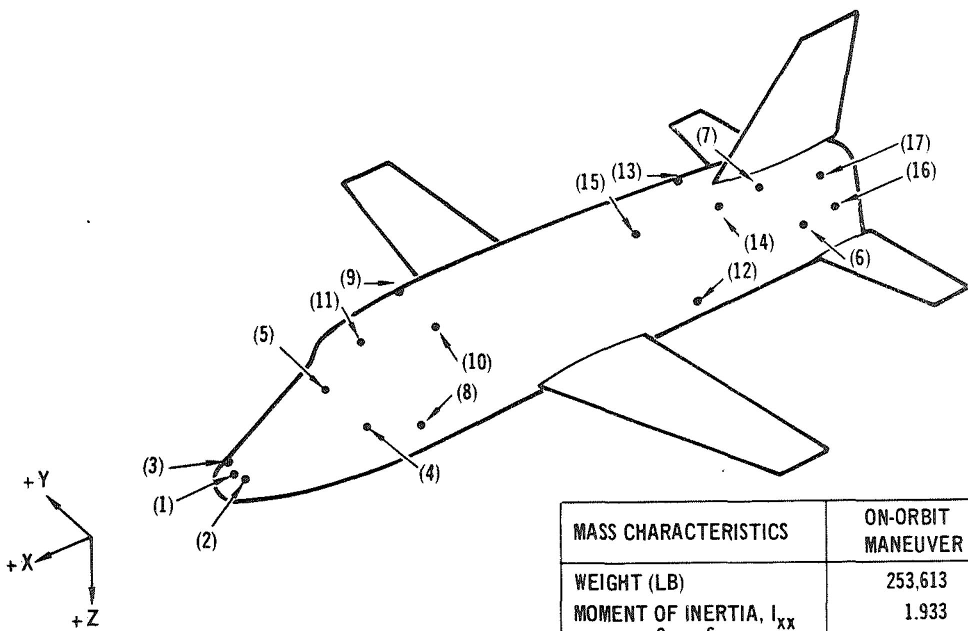
* CRITICAL ACCELERATION

ATTITUDE CONTROL ACCELERATIONS – BOOSTER
Twenty 2500 Lb Thrust Engines

FIGURE 3-6

ment using this revised engine arrangement and the SSVDRD requirements is 864,000 lb-sec. This total impulse level can be satisfied by residual liquid and vapor propellants in the main engine tanks (as discussed later in Section 4) if residual hydrogen vapor temperature is reduced from the SSVDRD value of 450 R to 260 R. A method of achieving this reduction with a main engine feed line heat exchanger is discussed in Paragraph 4.1.

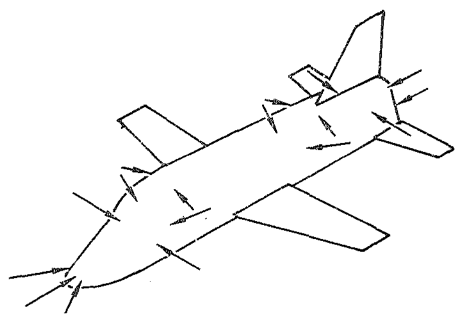
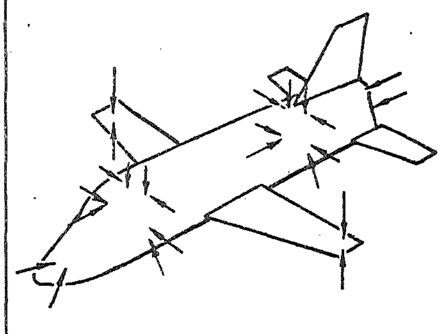
3.2 Orbiter - Orbiter APS mission requirements were met with an engine arrangement utilizing thirty-three 1080 lb thrust engines. The basic arrangement is illustrated in Figure 3-7. This arrangement also differs from the engine locations described in the SSVDRD which were nonoptimum for a low pressure APS. Several alternates were considered for the orbiter. An arrangement which minimized number of engines, thrust level, and associated supply line length proved to be most advantageous. This configuration, which is completely fuselage mounted, resulted in yaw and roll axes impulse penalties during entry. However, the hardware weight decreases which were achieved by eliminating wing-tip engines, far outweighed these penalties. Figure 3-8 summarizes incremental weight changes associated with the revised orbiter engine locations (compared to the configuration defined in Reference a). As shown, selected engine locations resulted in an 861 lb weight saving.



MASS CHARACTERISTICS	ON-ORBIT MANEUVER
WEIGHT (LB)	253,613
MOMENT OF INERTIA, I_{xx}	1.933
(SLUG-FT^2 x 10^6) I_{yy}	14.177
I_{zz}	14.559

ORBITER ENGINE ARRANGEMENT

FIGURE 3-7

PARAMETER	REVISED	SSVDRD
		
NUMBER OF ENGINES	33	40
THRUST, LB	1080	880
TOTAL IMPULSE, LB-SEC	3.259×10^6	3.259×10^6
SUBSYSTEM OPTIMUMS		
CHAMBER PRESSURE, LBF/IN^{2}A	13.9	13.6
MIXTURE RATIO	3	3
EXPANSION RATIO	8	8
SUBSYSTEM WEIGHTS, LB		
PROPELLANT (H_2)	1807	1761
(O_2)	5420	5284
TANK (H_2)	577	568
(O_2)	217	213
THERMAL CONDITIONING	567	558
DISTRIBUTION SYSTEM	1532	1798
ENGINES	2587	2786
(TOTAL)	(12707) (✓)	(13568)*

* INCLUDES 600 LB WEIGHT PENALTY FOR WING TIP PODS.

(✓) SELECTED

ORBITER ENGINE LOCATION COMPARISON

FIGURE 3-8

Figure 3-9 summarizes number of engines, their function, and installation data. Six engines were required for +X translation. Eight yaw engines are employed for on-orbit yaw and lateral maneuver requirements; they also provide primary yaw authority during reentry. These are backed up by canted pitch-roll engines. The pitch-roll engines would be activated if the differential between actual and commanded yaw rates exceeded predetermined values.

Selected engine arrangement capability is compared with SSVDRD requirements in Figures 3-10 and 3-11. Inspection of the figures shows that all attitude control accelerations and translational accelerations were achieved. Nominal acceleration levels are provided with one engine out, and minimal (safe) acceleration levels with two engines out, as required. Available reentry bank angle acceleration is 1.89 deg/sec^2, which exceeds the nominal minimum requirement of 1.5.

LOCATION	MANEUVER	NO. OF ENGINES	ENGINE COORDINATES* X	Y	Z	DIRECTION COSINE X	Y	Z
1	–X TRANSLATION	1	–260	0	–300	–1.0	0	0
2	–X TRANSLATION	1	–296	–45	–300	–0.926	+0.374	0
3	–X TRANSLATION	1	–296	+45	–300	–0.926	–0.374	0
4	+YAW, +Y TRANSLATION	2	–564	–73	–347	0	+1.0	0
5	–YAW, –Y TRANSLATION	2	–564	+73	–347	0	–1.0	0
6	–YAW, +Y TRANSLATION	2	–1954	–80	–300	0	+1.0	0
7	+YAW, –Y TRANSLATION	2	–1954	+80	–300	0	–1.0	0
8	–PITCH, –ROLL,REENTRY+YAW	2	–653	–135	–203	0	+0.707	+0.707
9	+PITCH, – ROLL,REENTRY–YAW	2	–653	+135	–343	0	–0.707	–0.707
10	+PITCH, + ROLL,REENTRY+YAW	2	–653	–135	–343	0	+0.707	–0.707
11	–PITCH, + ROLL, REENTRY–YAW	2	–653	+135	–203	0	–0.707	+0.707
12	+PITCH,–ROLL, REENTRY–YAW	2	–1865	–110	–195	0	+0.707	+0.707
13	–PITCH, –ROLL, REENTRY+YAW	2	–1865	+110	–405	0	–0.707	–0.707
14	–PITXH,+ROLL, REENTRY–YAW	2	–1865	–110	–405	0	+0.707	–0.707
15	+PITCH,+ROLL, REENTRY+YAW	2	–1865	+110	–195	0	–0.707	+0.707
16	+ X TRANSLATION	3	–2180	–25	–188	+0.993	+0.027	–0.114
17	+ X TRANSLATION	3	–2180	+25	–188	+0.993	–0.027	–0.114

*SEE APPENDIX F FOR COORDINATE SYSTEM. ALL DIMENSIONS ARE IN INCHES.

ORBITER ENGINE LOCATIONS
Thirty-Three 1080 Lb Thrust Engines

FIGURE 3–9

AXIS	CONDITION	ON-ORBIT ACCELERATION (DEG/SEC^2) REQUIRED	MIN AVAILABLE	REENTRY ACCELERATION (DEG/SEC^2) REQUIRED	MIN AVAILABLE
±PITCH	ALL ENGINES OPERATING	–	1.140	–	1.326
	ONE RING OUT	0.50	0.856	1.00	1.00**
	TWO RINGS OUT	0.30	0.570	0.50	0.612
+YAW*	ALL ENGINES OPERATING	–	0.928	–	2.330
	ONE ENGINE OUT	0.50	0.696	1.30	2.052
	(ONE RING OUT)			(1.30)	(1.984)
	TWO ENGINES OUT	0.30	0.464	0.90	1.774
	(TWO RINGS OUT)			(0.90)	(1.636)
±ROLL	ALL ENGINES OPERATING	–	3.050	–	3.122
	ONE RING OUT	0.50	2.269	1.00	2.322
	TWO RINGS OUT	0.30	1.498	0.50	1.522

* YAW ENGINES BACKED UP BY PITCH-ROLL ENGINES DURING REENTRY
** CRITICAL ACCELERATION

ATTITUDE CONTROL ACCELERATIONS – ORBITER
Thirty-Three 1080 Lb Thrust Engines

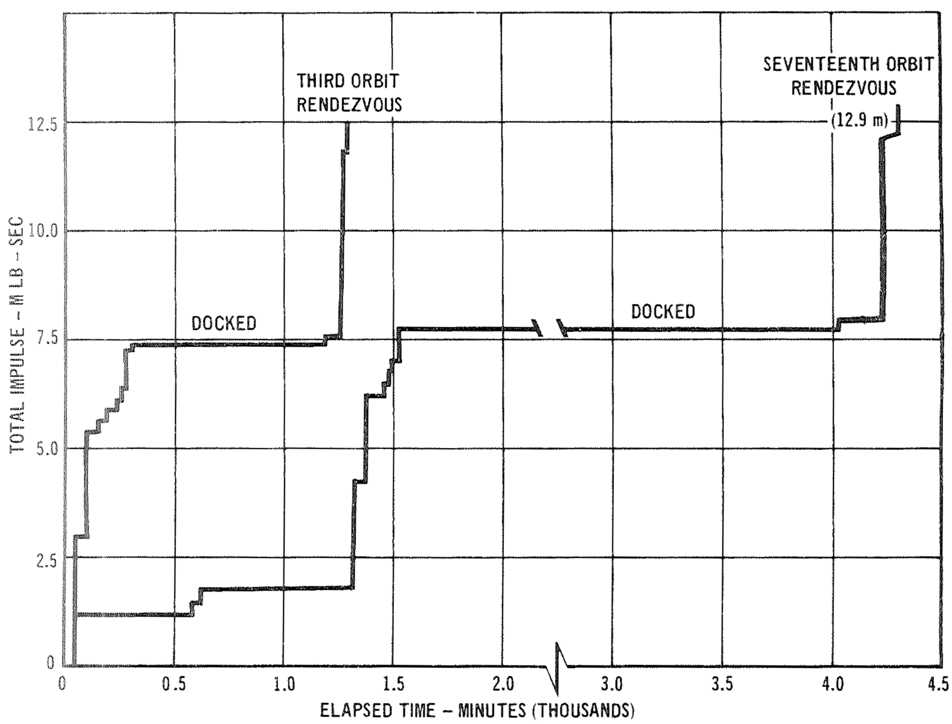
FIGURE 3–10

AXIS	CONDITION	TRANSLATIONAL ACCELERATION (FT/SEC^2)	
		REQUIRED	MIN AVAILABLE
+X	ALL ENGINES OPERATING	-	0.820
	ONE ENGINE OUT	0.65	0.684
	TWO ENGINES OUT	0.50	0.546
−X	ALL ENGINES OPERATING	-	0.381
	ONE ENGINE OUT	0.20	0.254
	TWO ENGINES OUT	0	0.085
±Y	ALL ENGINES OPERATING	-	0.549
	ONE ENGINE OUT	0.20	0.411
	TWO ENGINES OUT	0	0.275
±Z	ALL ENGINES OPERATING	-	0.775
	ONE ENGINE OUT	0.20	0.582
	TWO ENGINES OUT	0	0.388

TRANSLATIONAL ACCELERATIONS – ORBITER
Thirty-Three 1080 Lb Thrust Engines

FIGURE 3–11

For the engine locations described above, orbiter APS total impulse requirements and impulse usage history were defined for SSVDRD mission time lines. Results are shown in Figure 3-12 for the two basic third and seventeenth orbit rendezvous missions. Total impulse requirements are 12.5 and 12.9 M lb-sec, respectively.



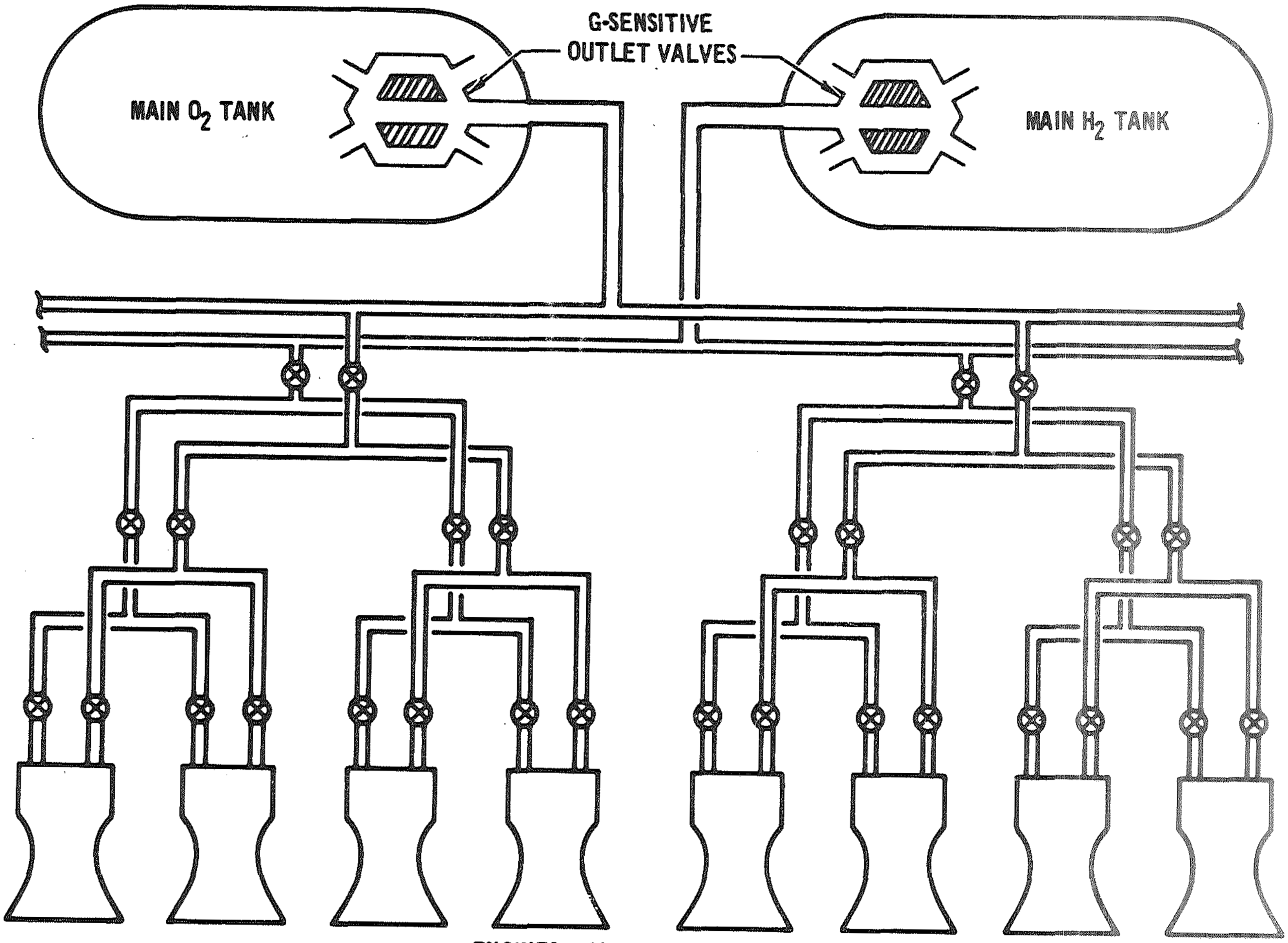
ORBITER IMPULSE TIME HISTORY
(SPACE STATION/BASE LOGISTICS MISSION)

FIGURE 3–12

4. BASELINE APS DESIGN

Final baseline designs resulting from the Subtask B study are discussed in the following paragraphs. Designs shown reflect requirements and engine arrangements presented in Section 3. Operational, performance, and component design characteristics are presented in Sections 5 and 6 along with discussion of alternates and selection justification.

4.1 Booster APS - The booster APS consists of propellant distribution, and engine assemblies, as shown schematically in Figure 4-1. The subsystem operates entirely from main engine tank vapors in a blowdown mode, over a pressure range of 26 to 17 lbf/in^2a. Acceptable propellant quality is ensured by g-sensitive liquid/vapor separators. The booster APS conforms fully to all SSVDRD requirements and design criteria, except for a reduction in hydrogen main engine tank vapor tempera-



BOOSTER BASELINE AUXILIARY PROPULSION SUBSYSTEM

FIGURE 4-1

ture at main engine shutdown. Temperature reduction can be readily accomplished (Figure 4-2) with passive heat exchangers, mounted to main engine hydrogen feed line, for pressurant conditioning during launch. In this manner, sufficient hydrogen residuals for full APS usage can be accommodated without effecting booster and orbiter main engine commonality, and without requiring separate APS propellant storage.

A summary of baseline design features is shown in Figure 4-3. The distribu-

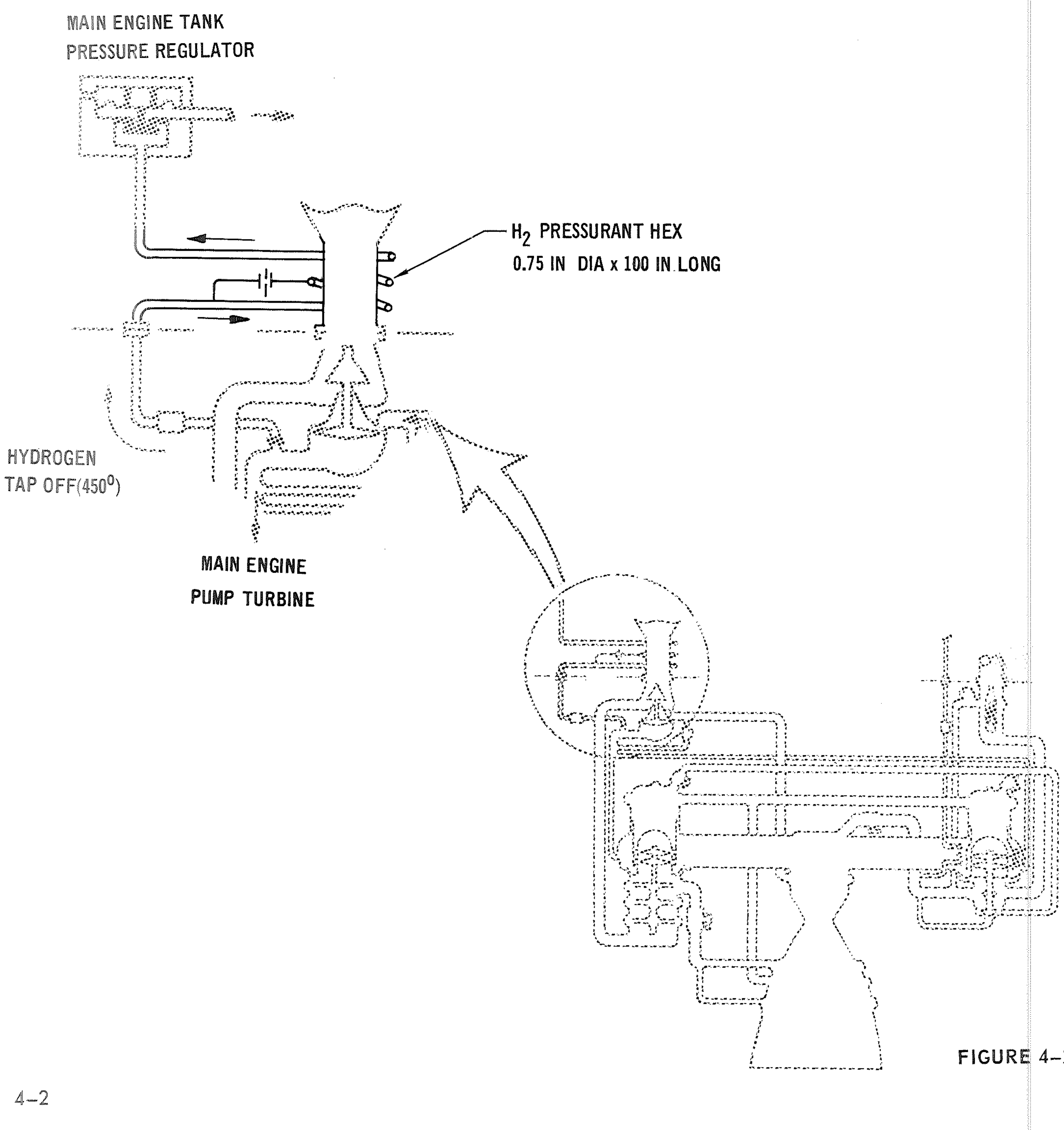


FIGURE 4-2

	O_2	H_2
MAIN TANK		
INITIAL VAPOR TEMPERATURE, ^oR	520	260
INITIAL PRESSURE, LBF/IN^{2}A	26	26
MINIMUM PRESSURE, LBF/IN^{2}A	17	17
ENGINE AND DISTRIBUTION SYSTEM		
DESIGN ENGINE INLET PRESSURE, LBF/IN^{2}A	14	14
DESIGN ENGINE INLET TEMPERATURE, ^oR	400	150
ENGINE THRUST, LB	2500	
MIXTURE RATIO	2.0	
CHAMBER PRESSURE, LBF/IN^{2}A	11	
EXPANSION RATIO	2:1	

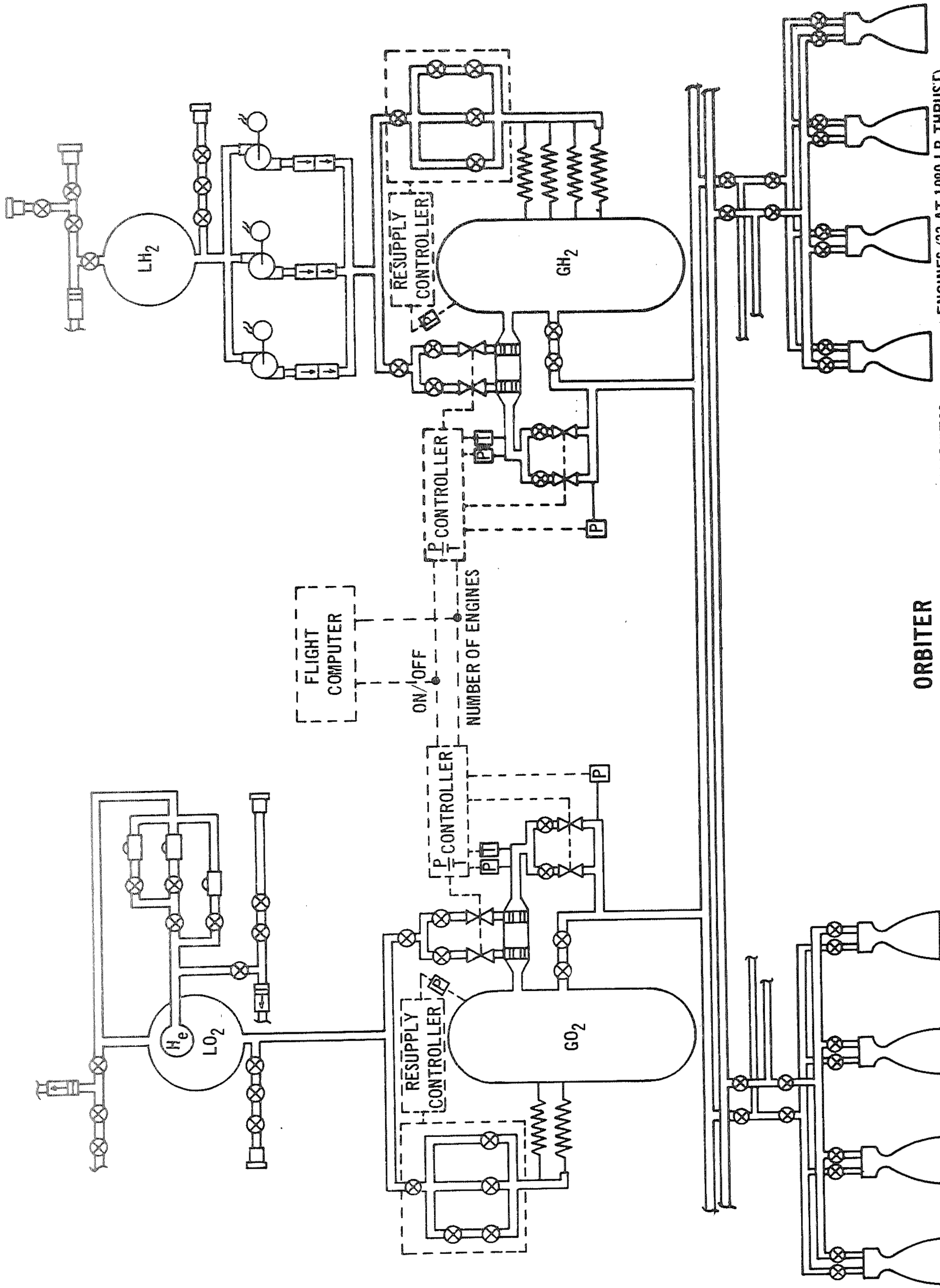
BOOSTER BASELINE DESIGN SUMMARY

FIGURE 4-3

tion and engine subsystem were sized to provide 2500 lb thrust at the end of blowdown, where inlet pressures and temperatures are at their lowest values. A mixture ratio of 2.0 and an expansion ratio of 2:1 were used for minimum subsystem weight. Twenty engines, arranged as shown in Appendix E, Figure E-7, provide required control accelerations. Subsystem installation takes advantage of main engine pressurant lines which extend nearly the full length of the vehicle; thus, required APS manifolding is minimal. All engines have been installed to be accessible from the exterior of the vehicle.

4.2 Orbiter APS - The orbiter APS consists of propellant storage, conditioning, control, distribution, and engine assemblies, shown schematically in Figure 4-4. The subsystem uses auxiliary liquid propellant storage; passive, tank-mounted, heat exchangers for main tank resupply conditioning; and liquid/vapor mixers to provide constant outlet density during major APS operations. Orbiter impulse requirements make a simple blowdown subsystem (as used on the booster) impractical. Some means of controlling main engine tank conditions, and an auxiliary propellant storage system, is required. Use of liquid/vapor mixers for control provides two primary advantages. First, variations in engine inlet conditions (flow rate, temperature, mixture ratio) are minimized, reducing engine development and design problems, and limiting subsystem performance variations. Second, increased burn duration capability is obtained. Main engine tank flow requirements are greatly reduced, since a large percentage of total engine flow is supplied as liquid to the liquid/vapor mixer inlet. This reduces main engine tank pressure decay during a burn, allowing increased burn durations.

Inherent subsystem simplicity is maintained by using passive, tank-mounted



ORBITER
BASELINE AUXILIARY PROPULSION SUBSYSTEM

FIGURE 4–4

heat exchangers, which utilize main tank heat capacity to vaporize and superheat the resupply propellant. During off periods, main engine tank temperature is restored by orbital heating. Mission interactions with subsystem performance have been minimized by decoupling the heat exchangers from the varying thermal environment of the vehicle skin.

Feed line weight is minimized by using existing main engine tank pressurization lines as primary APS distribution lines. These lines extend nearly the full length of the orbiter, and result in significant APS weight savings. The remainder of the distribution assembly has been sized to provide minimum subsystem weight by balancing line weight penalty as a function of frictional losses, and engine weight penalty as a function of chamber pressure.

Baseline characteristics are summarized in Figure 4-5 and subsystem installation is shown in Appendix E, Figure E-8. The APS provides all attitude control functions and all small maneuvers. Large, +X axis maneuvers are provided by a separate orbit maneuver subsystem (OMS) located in the aft portion of the vehicle. The OMS provides a maneuvering capability of 1150 ft/sec, covering three or four large burns. The APS provides the remainder of the maneuvers, which consist of burns less than 40 ft/sec. The resulting APS impulse requirement is 3.345 (10^6) lb-sec.

	O_2	H_2
LIQUID STORAGE AND PRESSURIZATION ASSEMBLY		
PROPELLANT WEIGHT, LB	4496	2499
PROPELLANT TANK PRESSURE, LBF/IN^2A	35	40
PROPELLANT TANK VOLUME, FT^3	67	634
PRESSURIZATION TYPE	COLD H_e	PUMP
PROPELLANT SUPPLY PRESSURE, LBF/IN^2A	35	35
HEAT EXCHANGER		
AREA, FT^2	1,790	3,100
TUBE LENGTH, FT	17.5	15.0
TUBE SPACING, IN.	4.0	10.0
TUBE DIAMETER, IN.	0.394	0.298
NUMBER OF TUBES	308	248
LIQUID-VAPOR MIXER		
OUTLET TEMPERATURE, ^oR	200	150
INJECTOR INLET PRESSURE, LBF/IN^2A	30	30
LIQUID THROTTLE RATIO	10:1	10:1
GAS-SIDE PRESSURE DROP, LBF/IN^2D	1.0	1.5
ENGINE AND DISTRIBUTION ASSEMBLIES		
DESIGN REGULATED PRESSURE, LBF/IN^2A	20	20
MAXIMUM LINE DIAMETER, IN.	8.3	8.3
ENGINE INLET PRESSURE, LBF/IN^2A	15.7	15.7
ENGINE THRUST, LB	1,080	
CHAMBER PRESSURE, LBF/IN^2A	13.7	
MIXTURE RATIO	3:1	
EXPANSION RATIO	8:1	

ORBITER BASELINE DESIGN SUMMARY

FIGURE 4-5

5. SUBSYSTEM OPERATION AND PERFORMANCE

This section describes subsystem operation and performance, first in terms of individual component characteristics, then in terms of the integrated subsystem operating in a mission duty cycle. Finally, transient performance characteristics are defined.

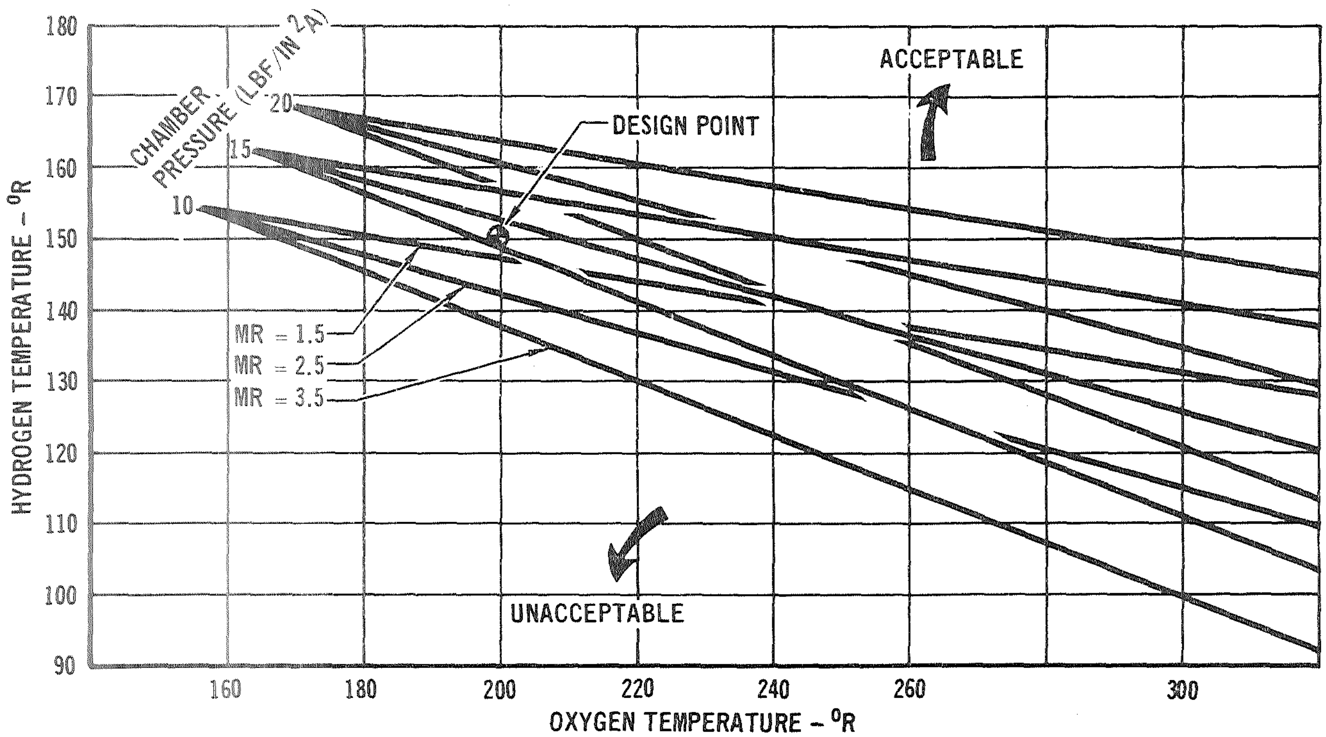
5.1 Component Performance Characteristics - The most significant components, the engine, heat exchanger (orbiter), and liquid/vapor mixer (orbiter) received the majority of study effort since their performance strongly influences total APS performance and has a large impact on subsystem weight. The remaining components, such as the propellant tankage are significant in terms of weight or technology but do not appreciably effect performance. Design of all components is discussed in Section 6.

(a) Engine - The engine is a film cooled design, using multiple coaxial injection elements and pilot operated, coaxial propellant valves. Booster and orbiter engines employ the same basic design concept, but have been sized to provide specific thrust requirements at chamber pressures, expansion ratios, and mixture ratios which result in minimum weight for each subsystem. The resulting booster engines are designed to provide 2500 lb thrust at a mixture ratio of 2.0 and expansion ratio of 2:1. Orbiter engines provide 1080 lb thrust at a mixture ratio of 3.0 and an expansion ratio of 8:1. Baseline thruster design characteristics are summarized in Figure 5-1. Minimum engine inlet temperatures were established at the level required to prevent oxygen condensation caused by heat transfer between propellants. Required temperatures are shown in Figure 5-2 as a function of chamber pressure and mixture ratio. Selected minimum temperatures were 150°R

		ORBITER	BOOSTER
THRUST	– LB	1080	2500
CHAMBER PRESSURE	– LBF/IN.^{2}A	13.7	11.0
MIXTURE RATIO		3.0	2.0
EXPANSION RATIO		8:1	2:1
INLET TEMPERATURE	– O_2/H_2 – °R	200/150	400/150
SPECIFIC IMPULSE	– SEC	376.5	342
FLOW RATE (TOTAL)	– LB/SEC	2.87	7.31
FUEL FILM COOLANT	– PERCENT	15	10
WEIGHT – TOTAL	– LB	77.0	149.0

APS ENGINE DESIGN SUMMARY

FIGURE 5-1

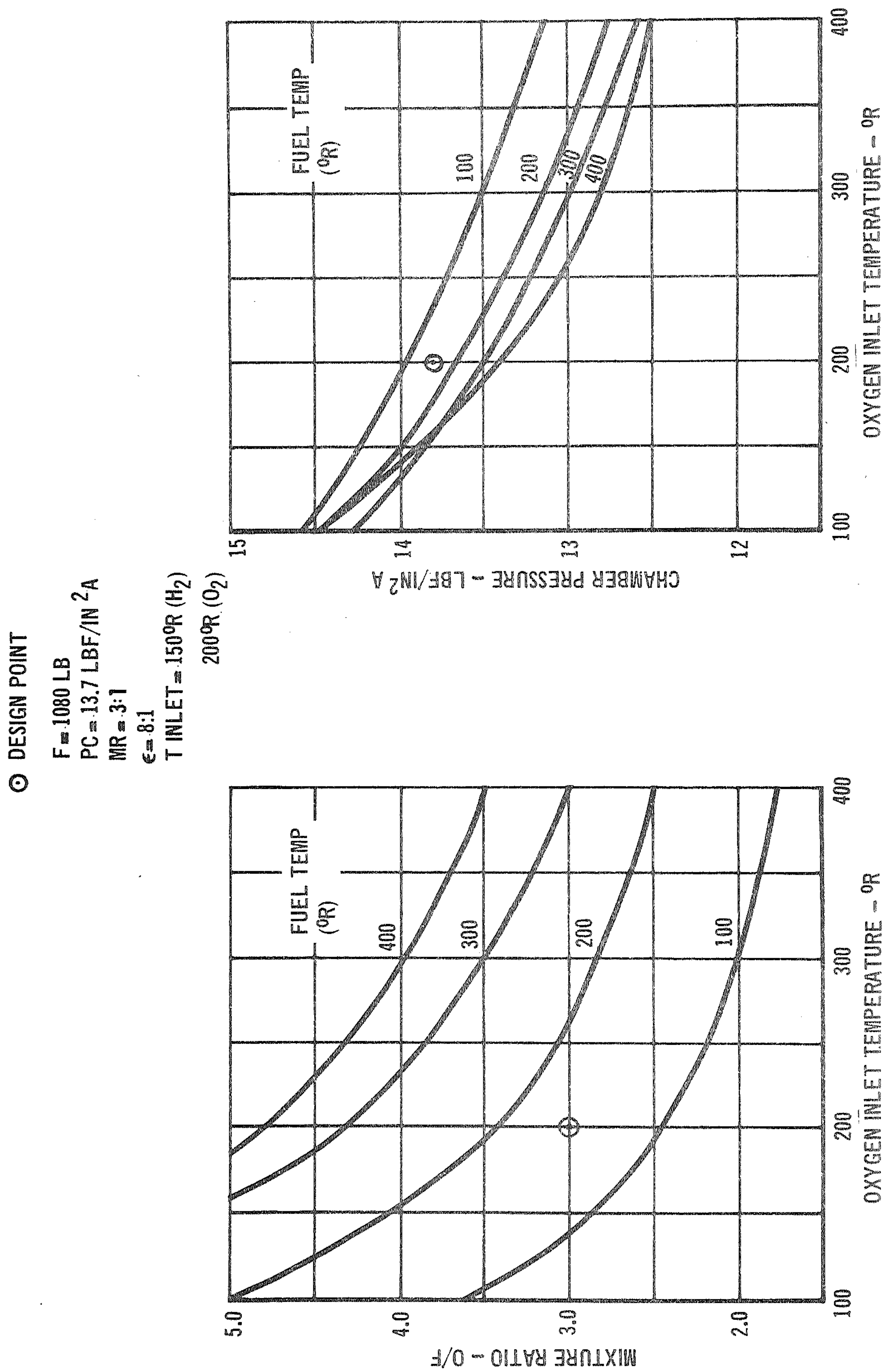


MINIMUM INJECTOR INLET TEMPERATURES

FIGURE 5-2

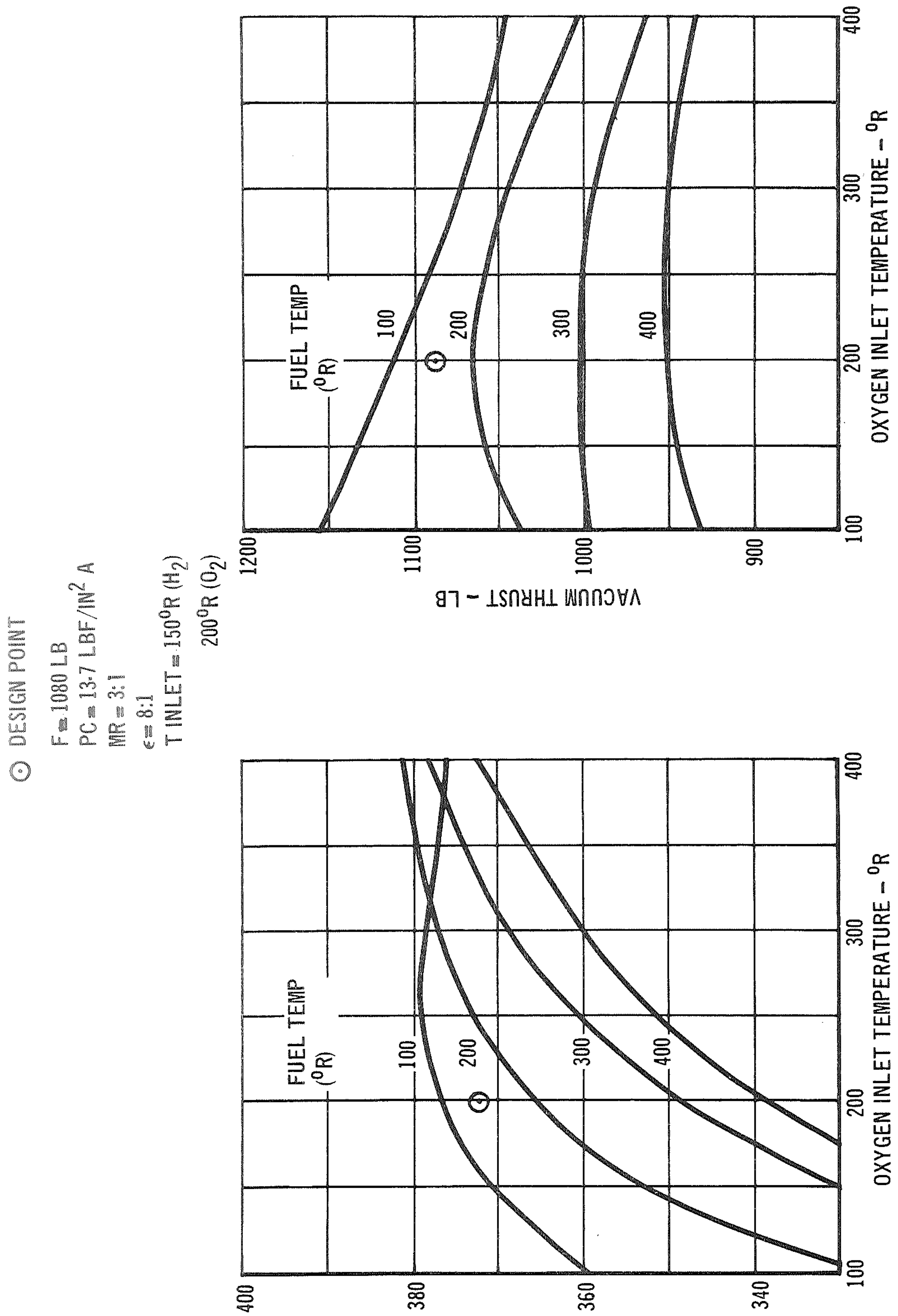
(hydrogen) and 200°R (oxygen). Figures 5-3 through 5-6 present orbiter engine design performance sensitivity (MR, Pc, F, Isp) to inlet temperature and inlet pressure variations and Figures 5-7 through 5-10 present booster performance sensitivity. Large performance variations result from opposing changes in hydrogen and oxygen temperatures and pressures. However, during a mission duty cycle, fuel and oxygen inlet conditions tend to vary in the same direction, which greatly reduces their effect on engine performance. As an example, mixture ratio variation from nominal design point is approximately ±1.0 for an opposing variation in inlet pressures of ±5 lbf/in^2. No significant mixture ratio variation occurs for a like pressure variation of ±5 lbf/in^2.

(b) Heat Exchangers (orbiter only) - The orbiter heat exchangers consist of tubes mounted directly to main engine propellant tankage. The heat exchanger operates essentially as a heat sink, absorbing heat from the propellant tank wall. During off periods, tank temperatures are restored by solar heating of the vehicle skin and reradiation to the tank. Since tank resupply (heat exchanger) flow rate is matched to tank outflow rate, the resupply enthalpy is directly proportional to the amount of heat extracted from the tank. As more heat is absorbed from the tank wall, resupply propellant temperature increases and final main tank vapor tempera-



ENGINE PERFORMANCE SENSITIVITY TO INLET TEMPERATURE
(ORBITER)

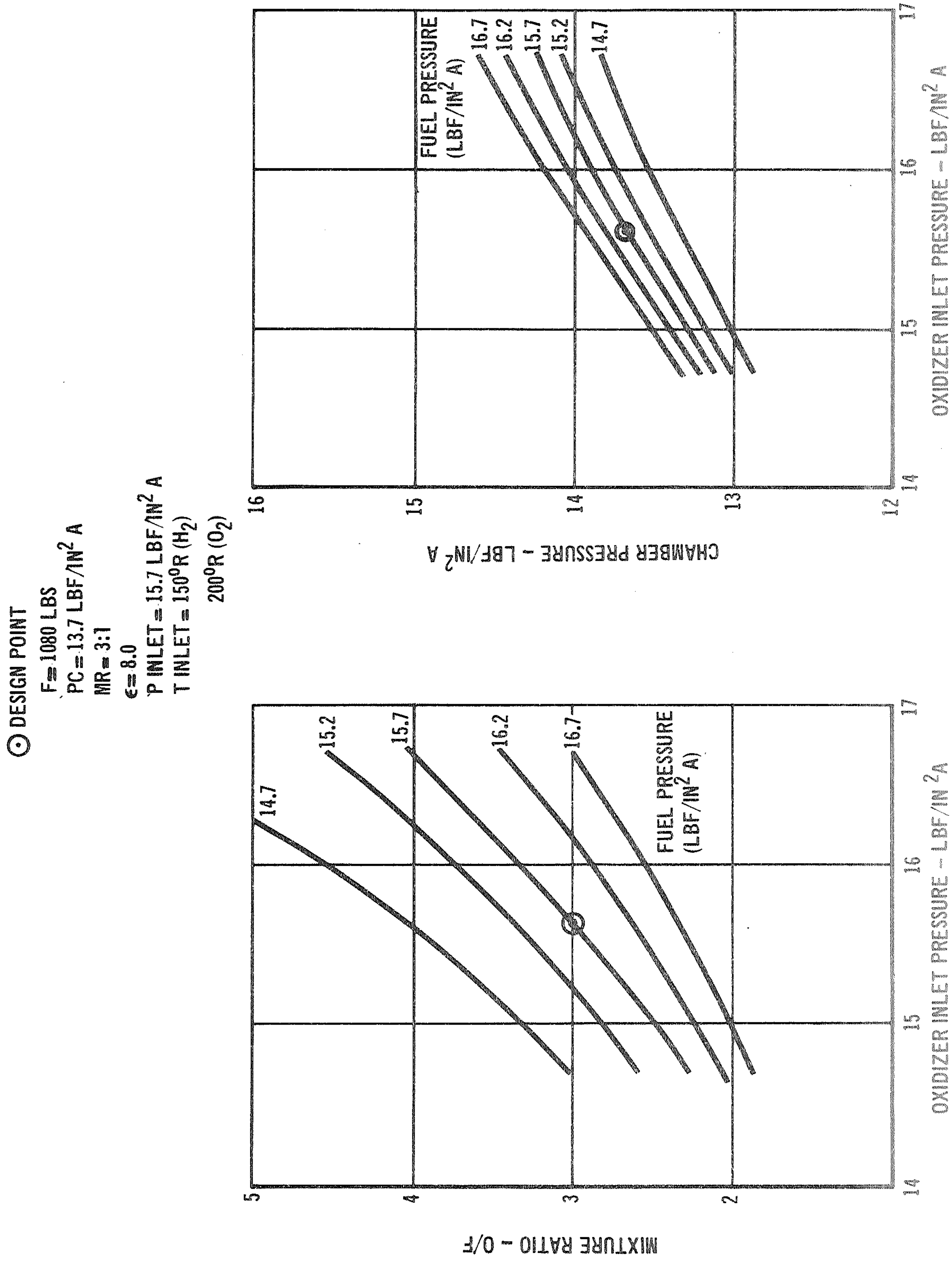
FIGURE 5–3



ENGINE PERFORMANCE SENSITIVITY TO INLET TEMPERATURE

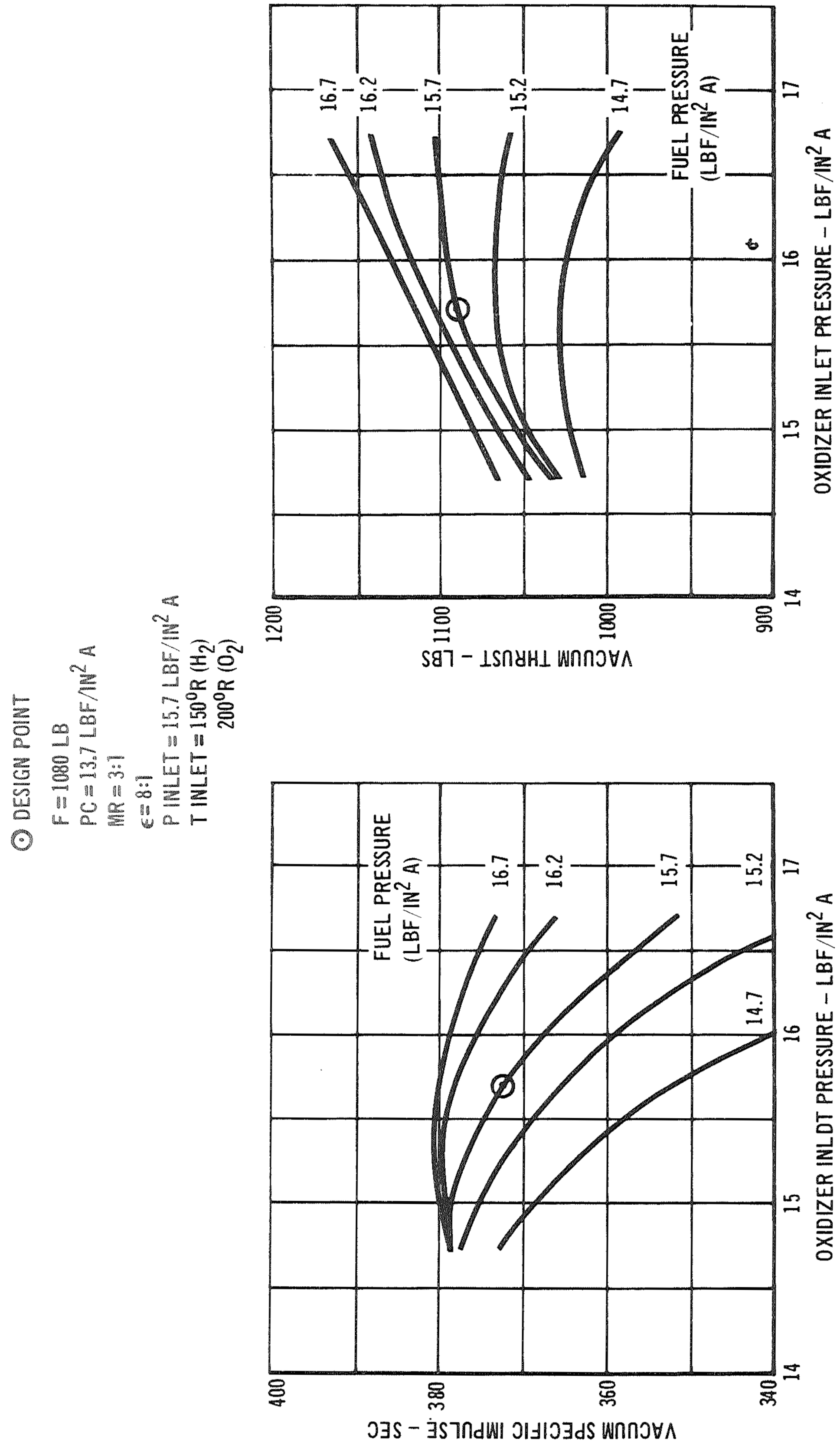
(ORBITER)

FIGURE 5–4



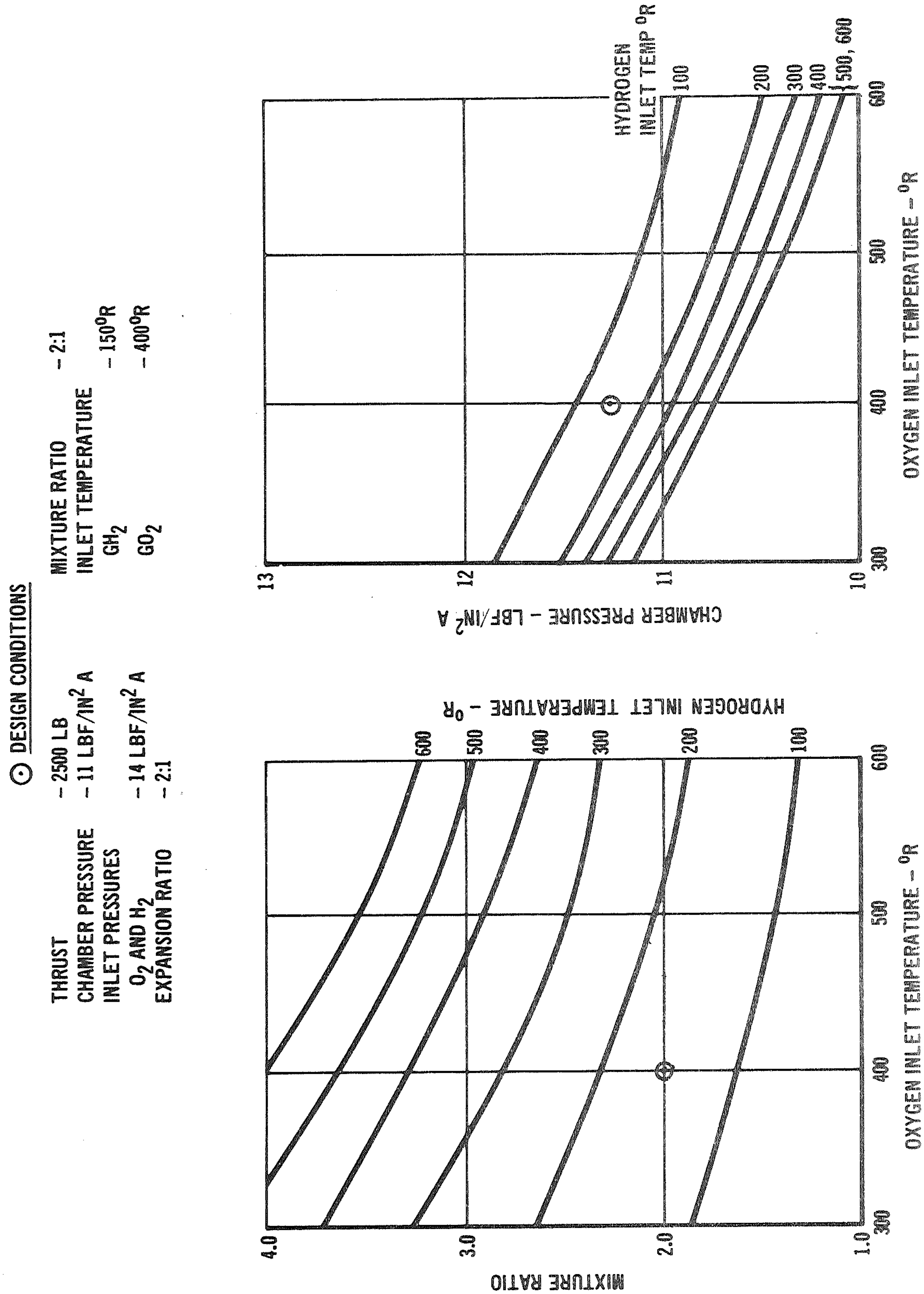
ENGINE PERFORMANCE SENSITIVITY TO INLET PRESSURE
(ORBITER)

FIGURE 5–5



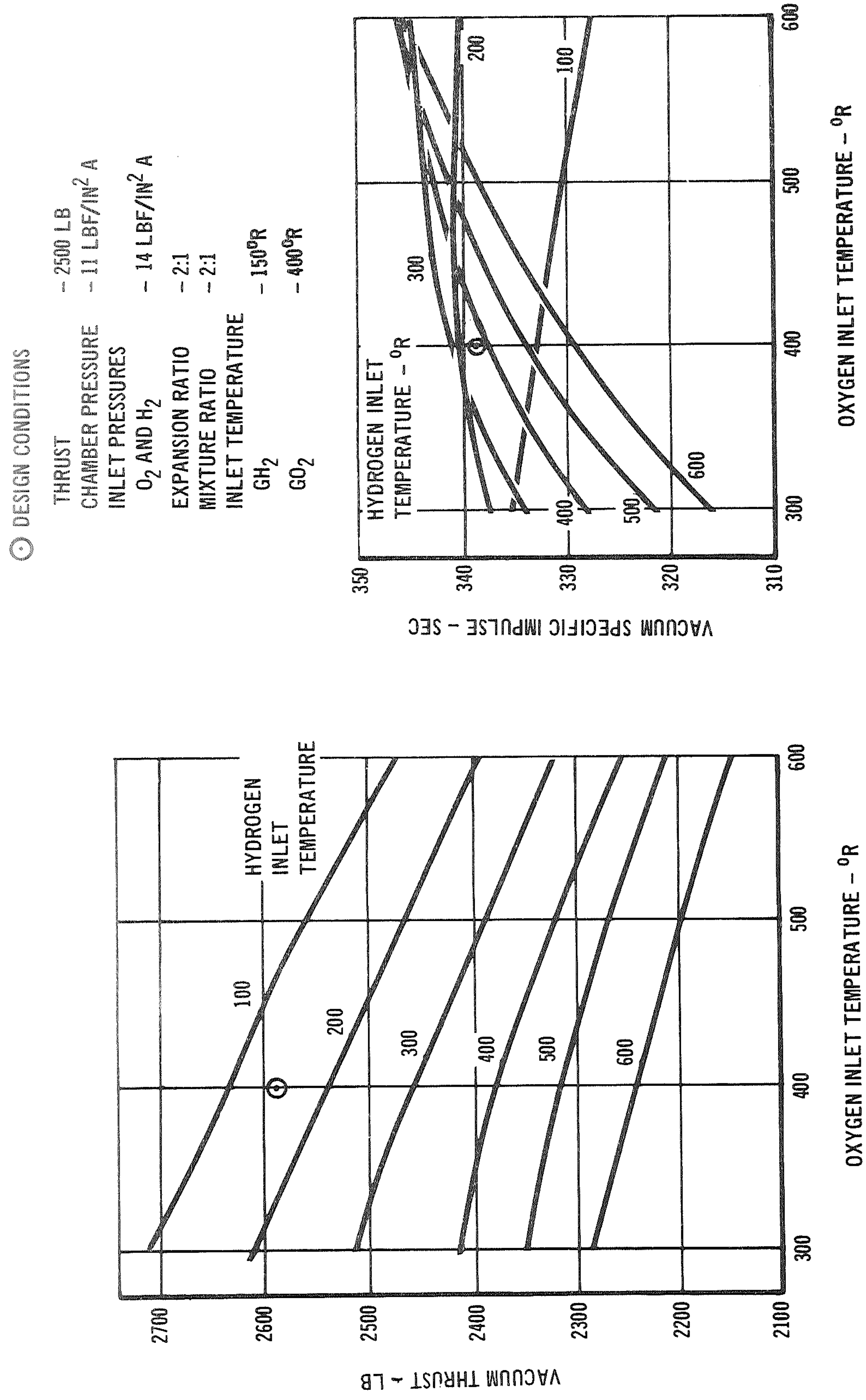
ENGINE PERFORMANCE SENSITIVITY TO INLET PRESSURE
(ORBITER)

FIGURE 5–6



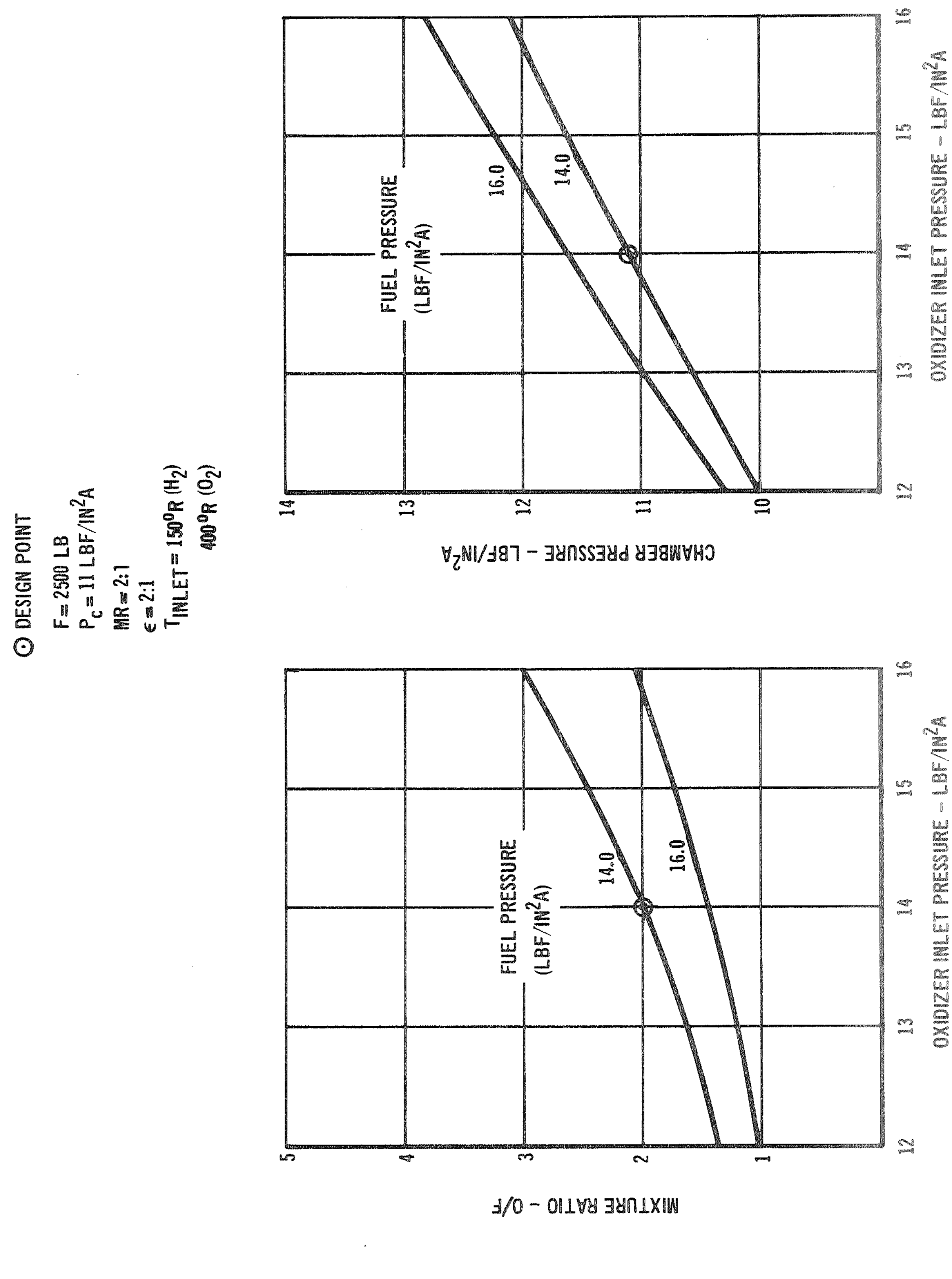
ENGINE PERFORMANCE SENSITIVITY TO INLET TEMPERATURE
(Booster)

FIGURE 5–7



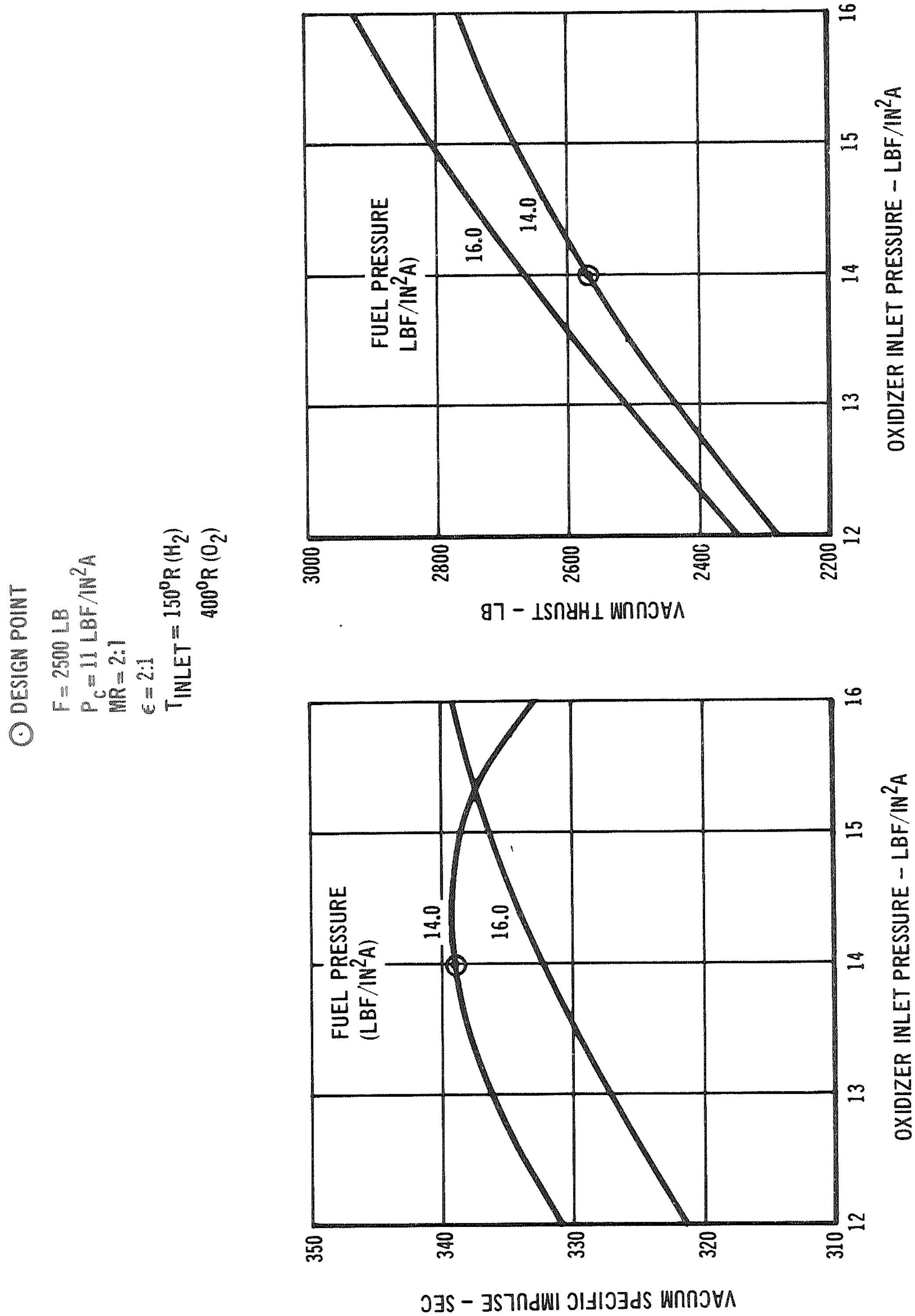
ENGINE PERFORMANCE SENSITIVITY TO INLET TEMPERATURE
(Booster)

FIGURE 5–8



ENGINE PERFORMANCE SENSITIVITY TO INLET PRESSURE
(Booster)

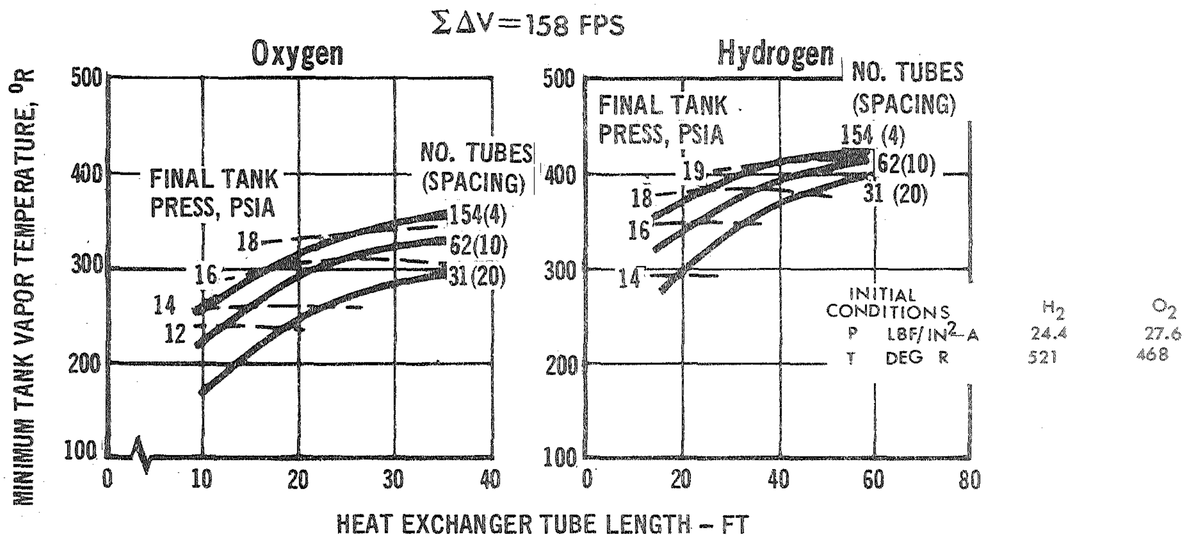
FIGURE 5–9



ENGINE PERFORMANCE SENSITIVITY TO INLET PRESSURE
(Booster)

FIGURE 5–10

ture and pressure increases. Therefore, heat exchanger design effectiveness can be measured by the resulting temperature and pressure of the vapor in the tanks after resupply operation. The effect of heat exchanger size (length, spacing, number of tubes) on main tank vapor properties are shown in Figure 5-11. Number of tubes,

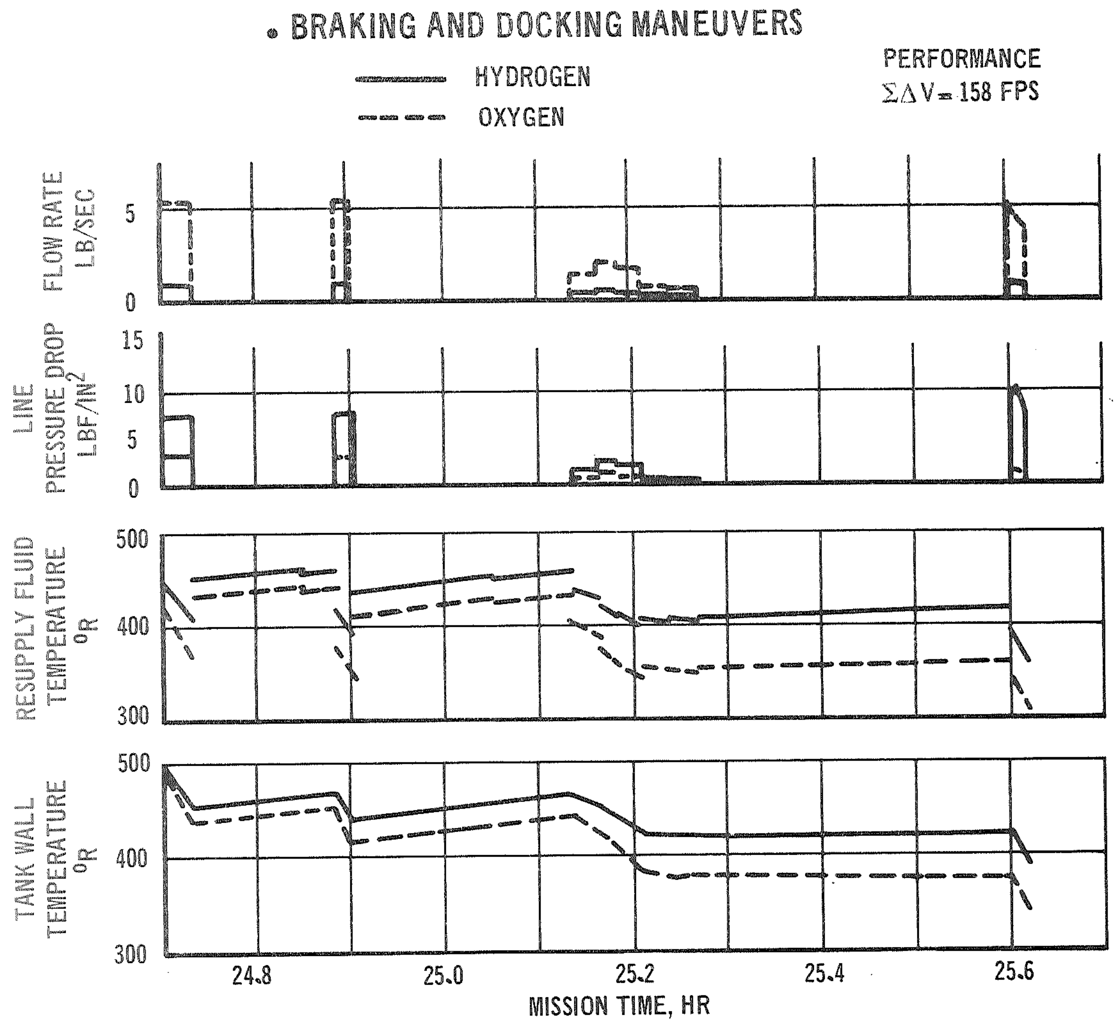


ORBITER MAIN TANK HEAT EXCHANGER

FIGURE 5-11

and tube spacing, reflects use of the entire tank perimeter. Thus, the length shown is the length of tank covered by the heat exchanger. The oxygen tank is smaller and has less available heat capacity, thus, the effect of heat exchanger characteristics on final vapor properties is more critical. For that reason, the oxygen heat exchanger covers the entire tank and a large number of closely spaced tubes are used (154 tubes at 4 in spacing). Further decreases in tube spacing would significantly increase number of tubes and heat exchanger weight, but would provide little improvement in final tank pressure or temperature. For the hydrogen heat exchanger, main engine tanks are larger and resupply flow rates are less, therefore heat exchanger design characteristics are less critical. Heat exchanger panel dimensions have been selected to give approximately the same final pressure as the oxygen tank. A heat exchanger design length of 60 feet was selected, with 62 tubes spaced 10 inches apart.

In order to reduce pressure drop encountered with long tubes, heat exchangers have been divided into parallel panels (two oxygen and four hydrogen). Although thermal performance is degraded since the heat exchangers are shorter, the resulting subsystem performance (shown in Figure 5-12) is good. Data shown are for the most critical mission phase, the period just prior to docking. During all other mission phases, resupply fluid and tank wall temperatures are significantly higher



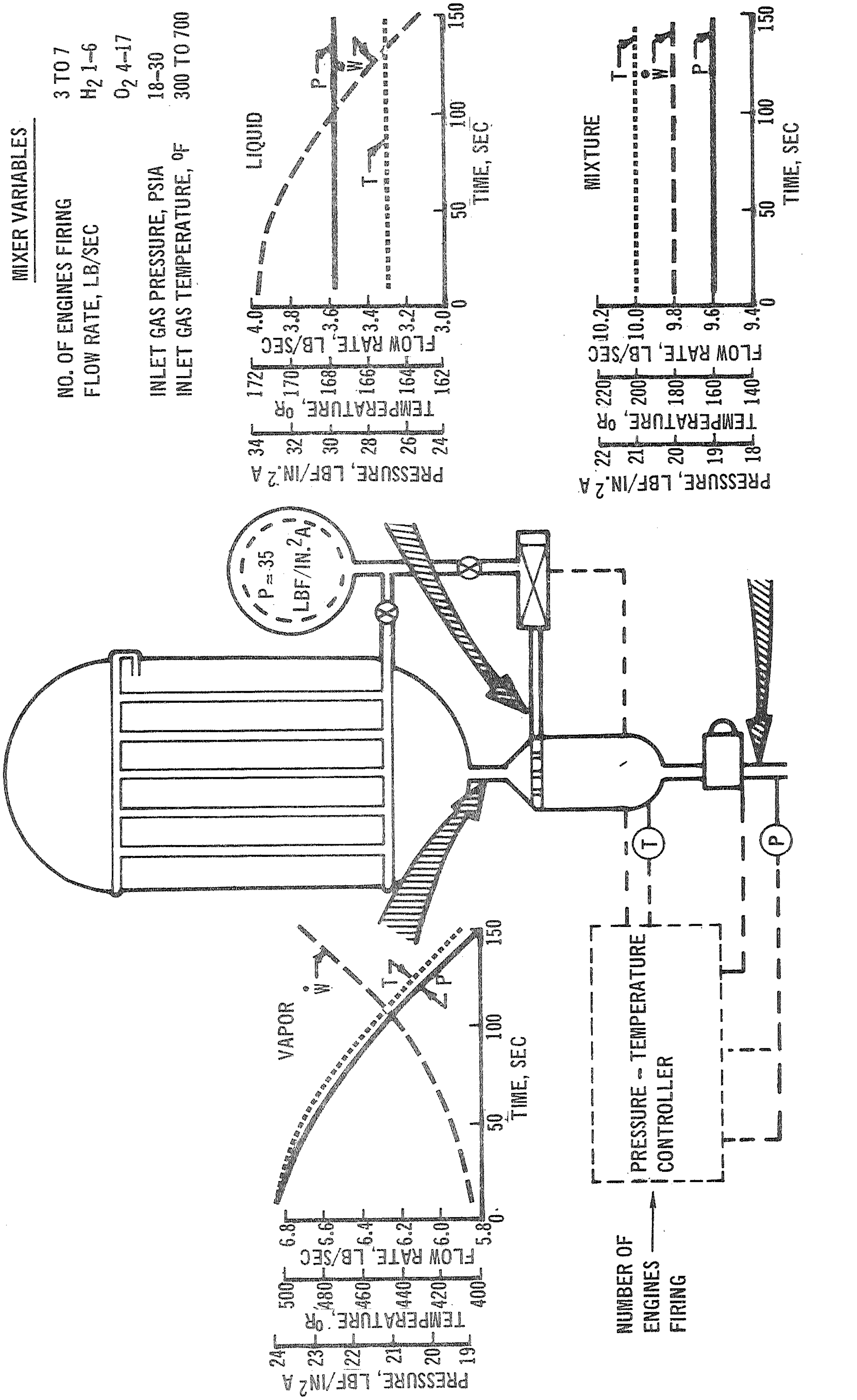
HEAT EXCHANGER OPERATION (SEVENTEENTH ORBIT RENDEZVOUS)
Braking and Docking Maneuvers

FIGURE 5-12

than the lowest values shown.

(c) Liquid/Vapor Mixers (Orbiter only) - The liquid-vapor mixer consists of injector and mixing chamber, liquid flow controller, and downstream regulator. During major APS operations, the mixer provides conditioned vapors at a prescribed temperature of 150°R (H_2) and 200°R (O_2) and an "Iris" valve controls downstream pressure to 19 lbf/in^{2}a. Liquid/vapor mixer operation is illustrated in Figure 5-13 for a typical entry mission phase. During operation, main engine tank pressure decays from 24 to 19 lbf/in^{2}a and vapor temperature decays from 500°R to 410°R. Associated with tank vapor temperature decay, the quantity of liquid that can be conditioned to a prescribed outlet temperature is reduced, and liquid flowrate must be throttled.

Cavitating venturis are used to decouple liquid propellant feed from down-



LIQUID – VAPOR MIXER

Steady State Operating Characteristics – Oxygen

FIGURE 5–13

stream (mixer) pressure variations. Pilot-input commands from the flight computer signal the number of engines firing. This information establishes a feedback control bias for valve positioning. The controller senses downstream pressure and commands an iris regulator setting, resulting in 19 lbf/in^2a outlet pressure.

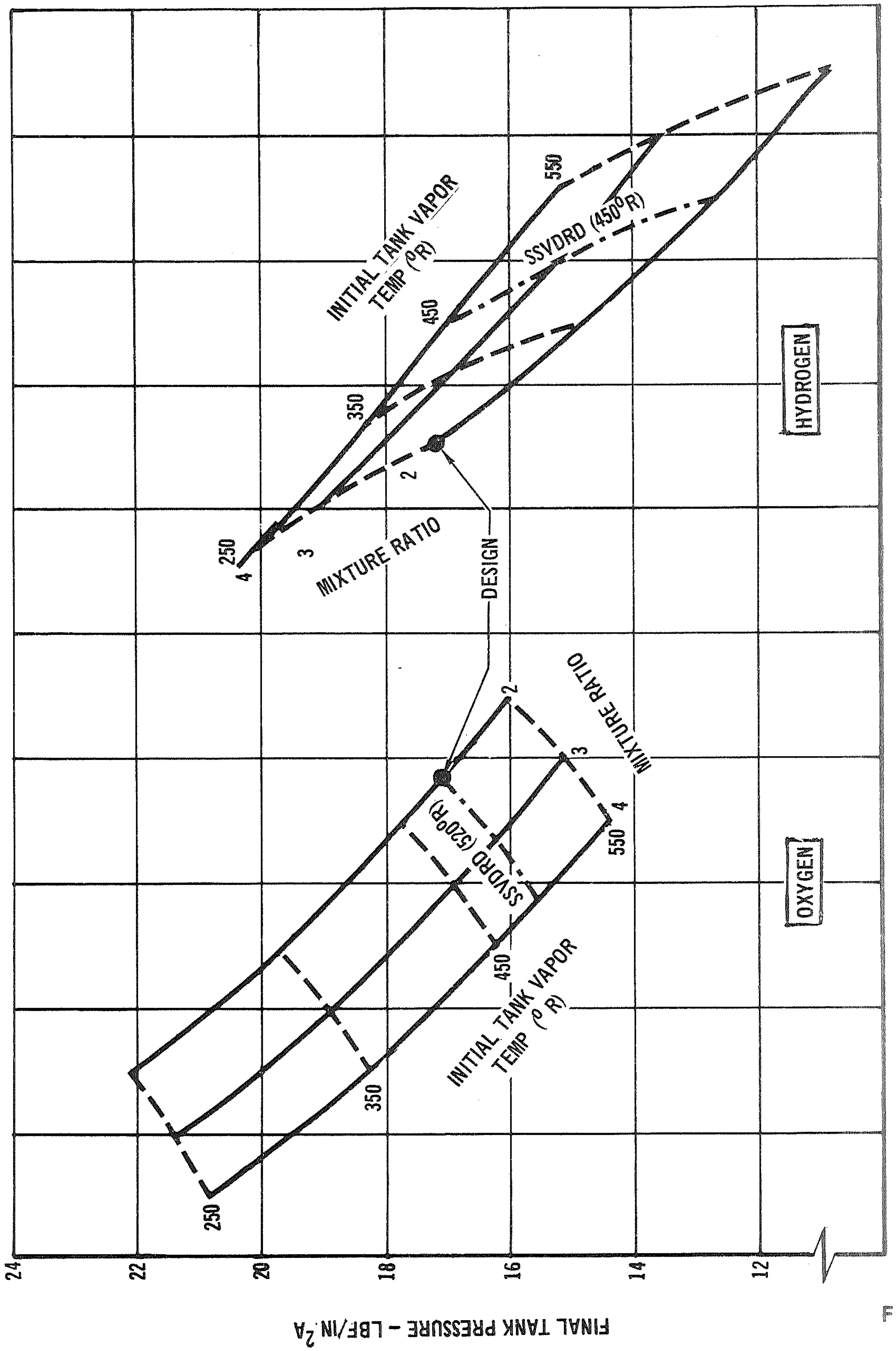
Liquid vapor mixer assembly is used in a dual-mode of operation, in which the mixer is utilized only during major maneuvers. During low APS usage, the iris regulator remains fixed at its last position and no liquid flow is provided to the mixer. Instead, main tank vapors provide all APS propellant. In this way, required component operating ranges have been minimized.

5.2 Mission Duty Cycle - Component performance characteristics cited above, and interactions between components, have been evaluated for both booster and orbiter. Resulting subsystem performance is presented below.

5.2.1 Booster Mission Duty Cycle - The booster APS utilizes a simple blowdown concept without auxiliary liquid propellant storage, resupply, or mixing. APS performance depends on initial propellant vapor conditions (those existing at end of main engine shutdown). Figure 5-14 shows the effect of initial tank vapor temperature on final tank pressure. For a mixture ratio of 2.0 and SSVDRD specified temperatures, final hydrogen pressure will be approximately 13 lbf/in^2a and final oxygen pressure will be 16 lbf/in^2a. With a mixture ratio of 3.5 the same final pressure (15.2 lbf/in^2a) would be obtained for each propellant. If lower initial vapor temperatures are provided, the amount of vapor at end of boost and final tank pressure will be increased accordingly. The effect of final tank pressure on subsystem weight is shown in Figure 5-15. These weights include the added vapor residuals attributable to reduced vapor temperatures. The SSVDRD conditions result in unnecessarily high weight penalties for the APS and, since pressurant gas temperatures could be relatively easily reduced by incorporating a heat exchanger (Section 4), lower initial vapor temperatures were selected as they were much more compatible for design with the low pressure APS.

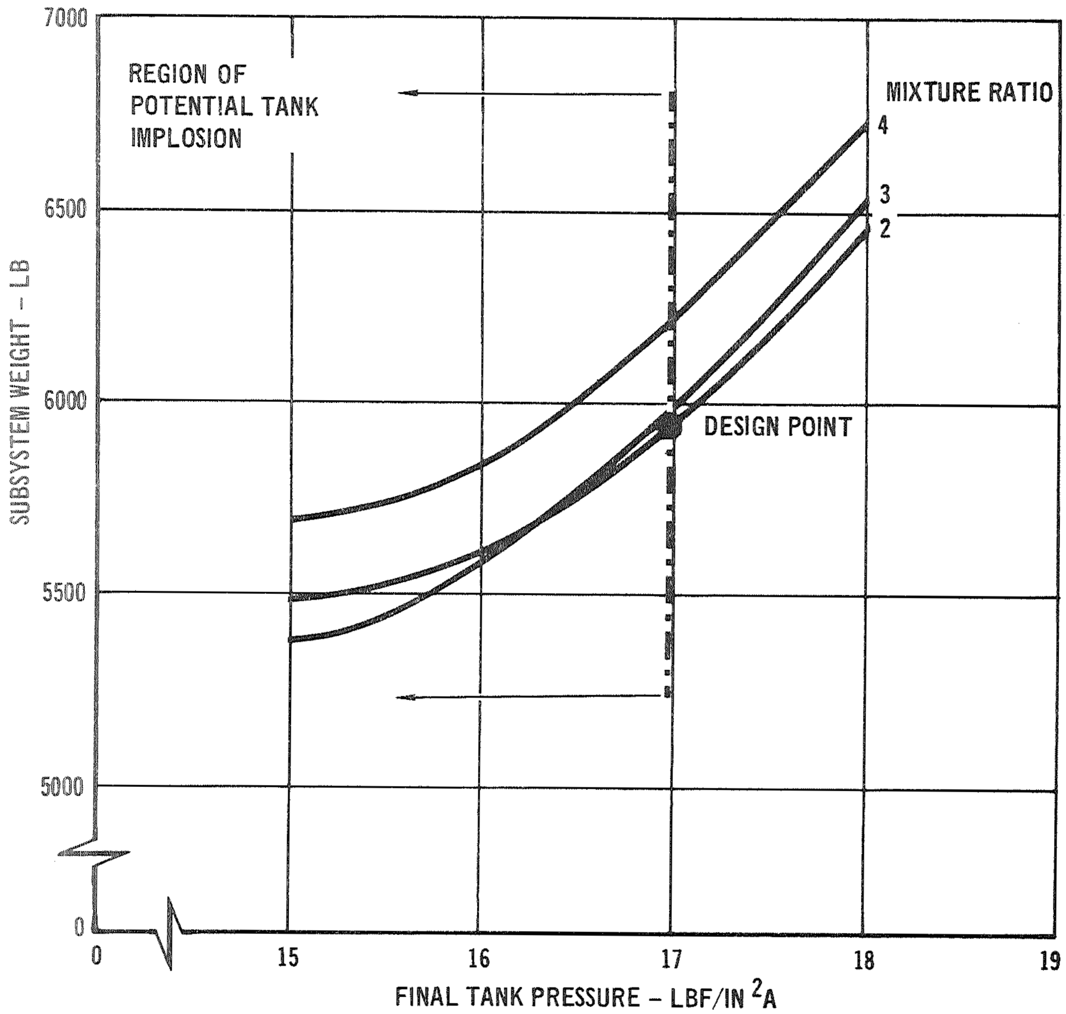
A design pressure level of 17 lbs/in^2a was selected to provide a positive pressure in the main tanks when in the atmospheric environments. Design vapor temperatures for this pressure are 260°R (H_2) and 520°R (O_2). An additional consideration in pressure level selection was engine ignition limitations. A minimum pressure of 2 to 3 lbf/in^2a is required in the chamber for reliable ignition. With 17 lbf/in^2a tank pressure, propellant cold flow results in a chamber pressure of approximately 5 lbf/in^2a, well above ignition limits.

Resultant booster tank pressure and temperature profiles during the mission



BOOSTER VAPOR TEMPERATURE – MIXTURE RATIO PARAMETRIC

FIGURE 5–14



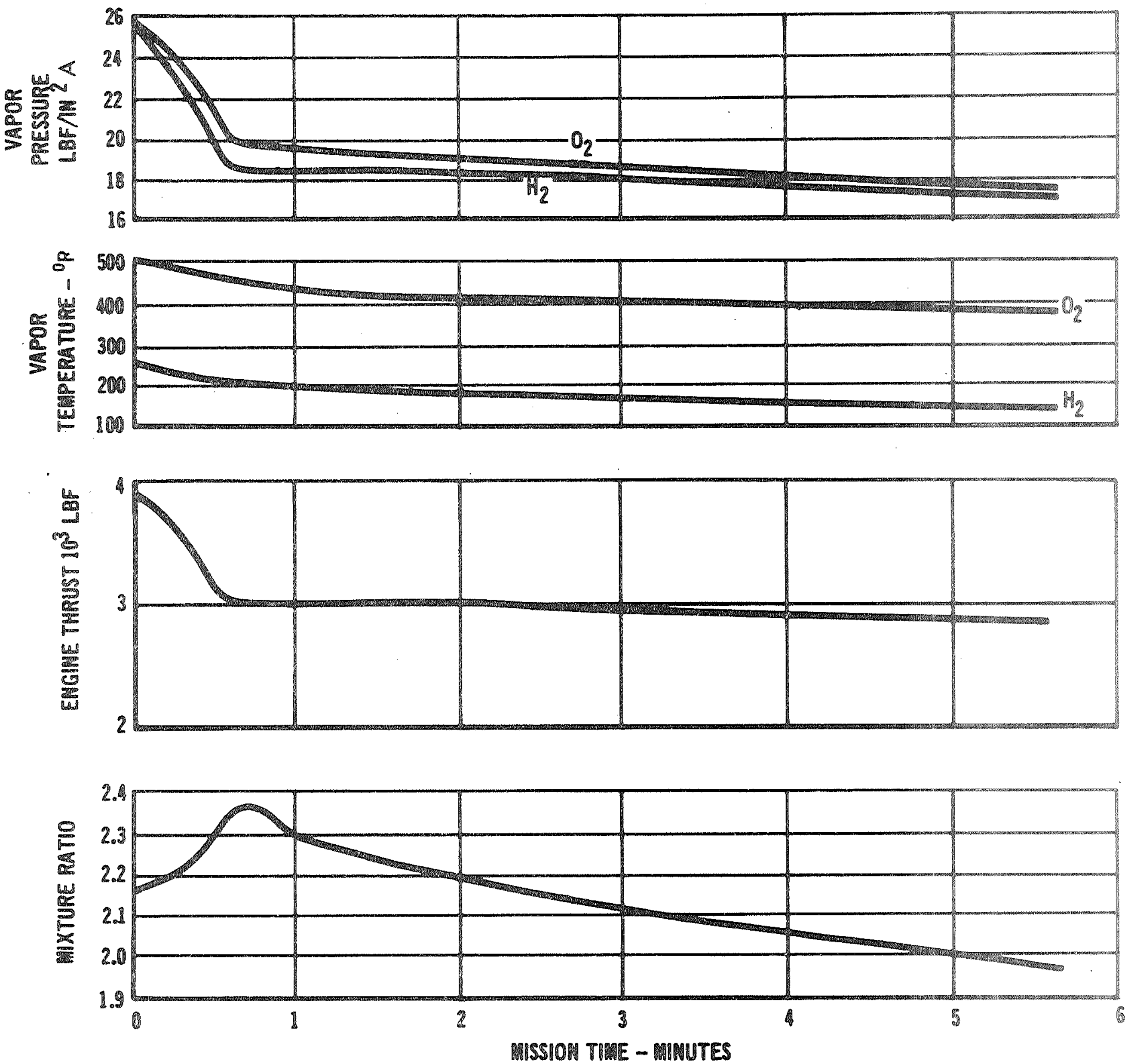
BOOSTER SUBSYSTEM WEIGHT PARAMETRIC

FIGURE 5–15

are shown in Figure 5-16. As shown, pressures and temperature variations result in only minor performance variations. Engine mixture ratio varies from 2.0 to 2.2, and thrust decays from 3800 to 2500 lb.

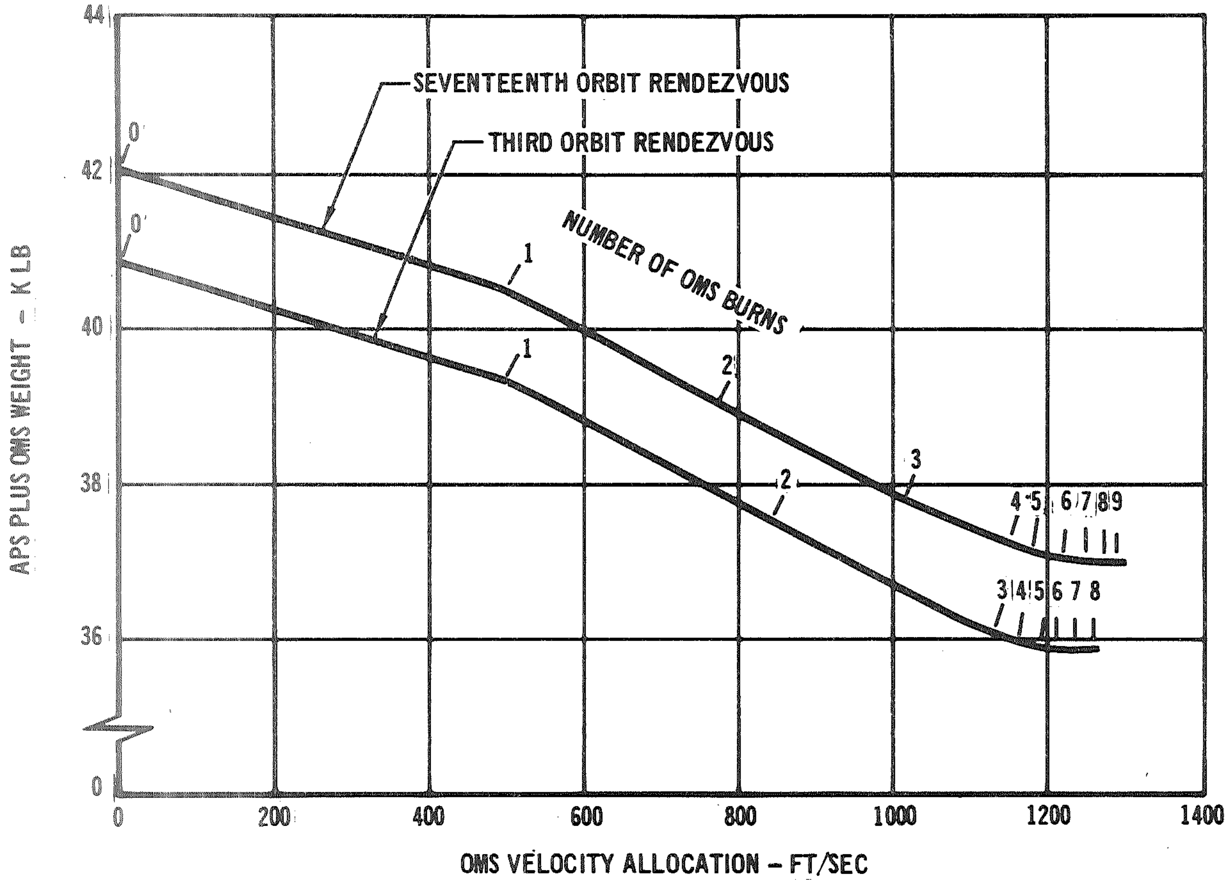
5.2.2 Orbiter Mission Duty Cycle - The orbiter subsystem operates in a blowdown mode (i.e., all propellant is extracted from main engine tanks and there is no downstream liquid injection) during periods when demand is low. When a major APS operation, such as a midcourse correction, is required, only part of the propellant is extracted from the main engine tank; the remainder is supplied as liquid to the mixing assembly, significantly reducing main engine tank pressure decay. Propellant, equal in quantity to that withdrawn, is resupplied to the main engine tank from the storage tank through the passive heat exchanger.

APS velocity allocations were based on a study to evaluate the optimal APS/OPM



BOOSTER APS MISSION OPERATING CHARACTERISTICS FIGURE 5-16

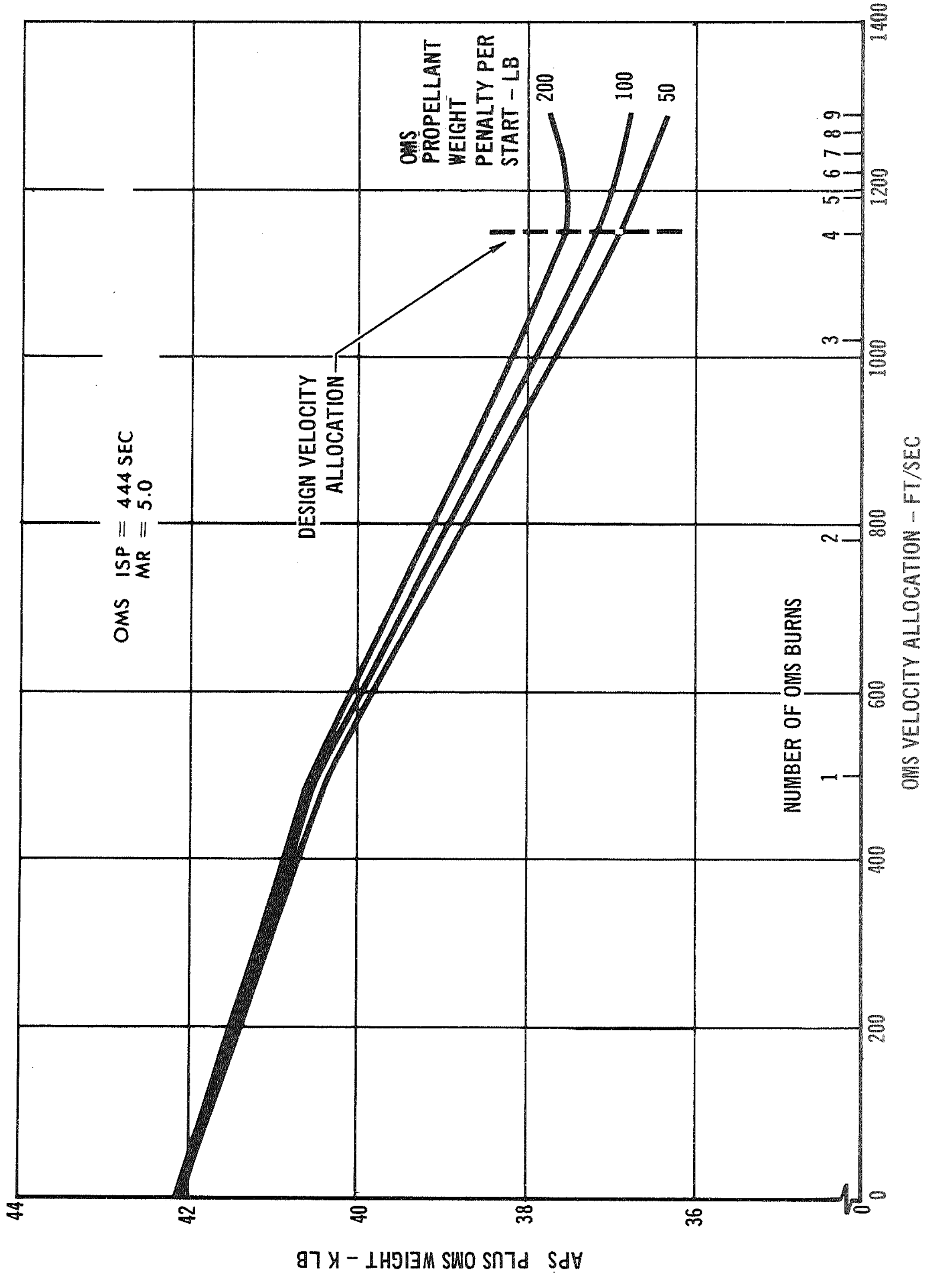
impulse split. Figure 5-17 presents study results. Shown are combined OMS and APS weights as a function of velocity allocated to the OMS. Both the third and seventeenth orbit rendezvous cases of Reference (b) were investigated. An RL-10-A3 OMS engine was assumed for this analysis. As shown in Figure 5-17, initially the combined weight of the two propulsion subsystems decreases markedly due to high OMS performance. This decrease in weight continues for the largest three or four mission maneuvers. These maneuvers are: deorbit burn, orbit height adjustment, coelliptic burn, and phasing burn. As illustrated, the number of OMS burns in-



APS/OMS IMPULSE ALLOCATION

FIGURE 5-17

creases very rapidly after these four major maneuvers, without appreciable decrease or increase in total OMS/APS weight. This results principally from increased start losses when the OMS engine is required to perform small maneuvers. Hence, an effective degradation of OMS specific impulse occurs. Inspection of Figure 5-17 shows that approximately 1150 ft/sec should be allocated to the OMS. This provides near minimum overall subsystem weight and only three to four starts on the liquid engine. In analyzing Figure 5-17, a major uncertainty is the amount of propellant required for each OMS engine start. Precise evaluation of this quantity requires detailed evaluation of OMS installation and was beyond the scope of this study. However, in order to assess the validity of conclusions drawn from Figure 5-17, OMS engine start losses were varied, and resulting combined subsystem weights were evaluated. In Figure 5-18, propellant losses for each OMS start were varied from 50 to 200 lb. As shown, at very low start losses no optimum impulse split occurs, but only a very small weight gain can be realized by increasing the number of OMS starts above four. Based on these results, it was concluded that an OMS velocity allocation of 1150 ft/sec was valid, independent of start loss.



APS/OMS IMPULSE ALLOCATION
Seventeenth Orbit Rendezvous

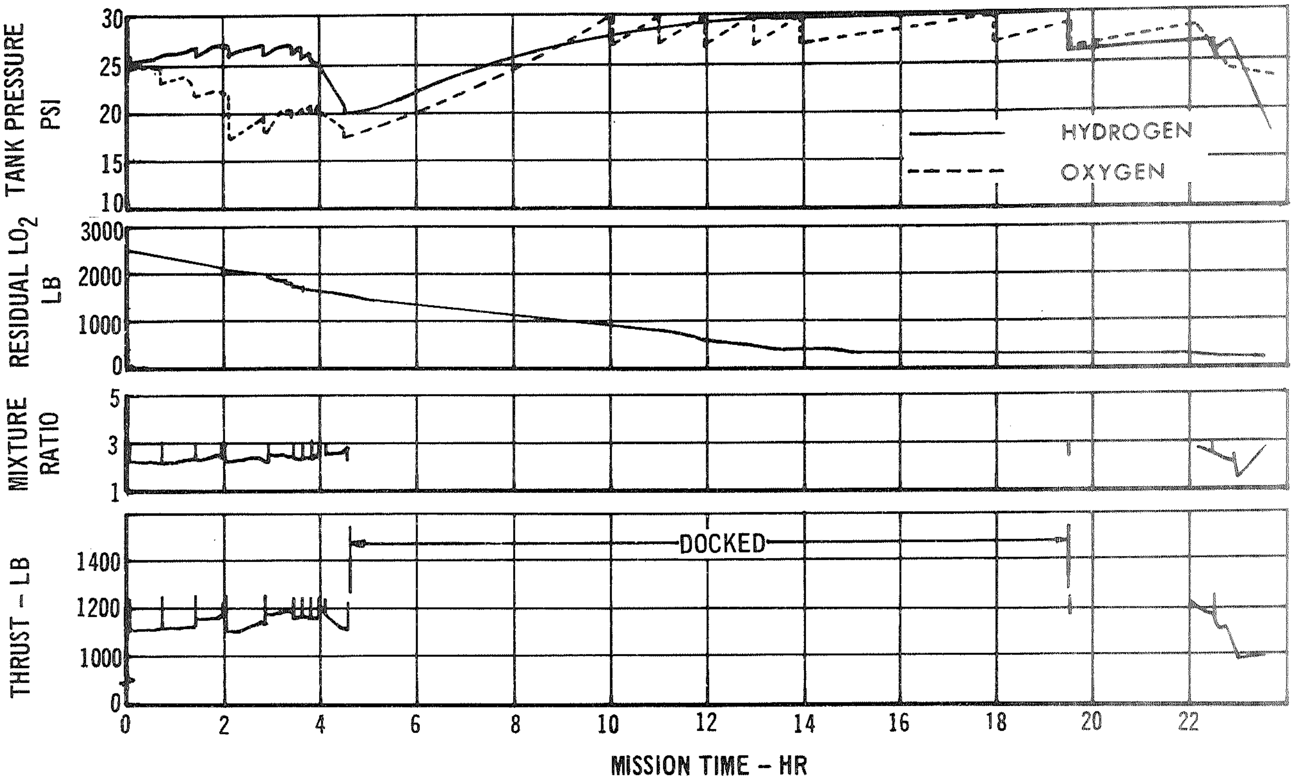
FIGURE 5–18

Mission profiles for low pressure APS are illustrated in Figure 5-19 and 5-20. At engine shutdown, main engine tanks contain residual propellants. Normal heat leak into the tank will warm the propellant vapor and boil off liquid residual. Without APS usage, tank pressure would increase and propellant would be vented. During the initial mission phase, the APS operates almost entirely from residual liquid boiloff in the main engine tanks. When (later in the mission) tank pressure has decayed and liquid residuals have been exhausted, main engine tank pressure is restored by resupplying conditioned propellant vapors.

During low demand attitude control operations, all propellant is supplied from the main engine tank. Engine inlet conditions are allowed to vary with tank temperature and pressure fluctuations. When a major APS operation is scheduled, only a part of the propellant is extracted from the tank, and engine inlet conditions are closely controlled by adding liquid to the mixer. After each major APS operation, propellant vapor in the main engine tank will be relatively cool and tank walls will have been chilled because of heat removal for propellant conditioning. Normal radiation from vehicle skin to tank walls will restore tank wall temperature which, in turn, will transfer heat to propellant vapors, raising gas temperature and pressure. Thermal interactions between tank and propellant vapors as well as the effect of environmental variations, are presented in Appendix B.

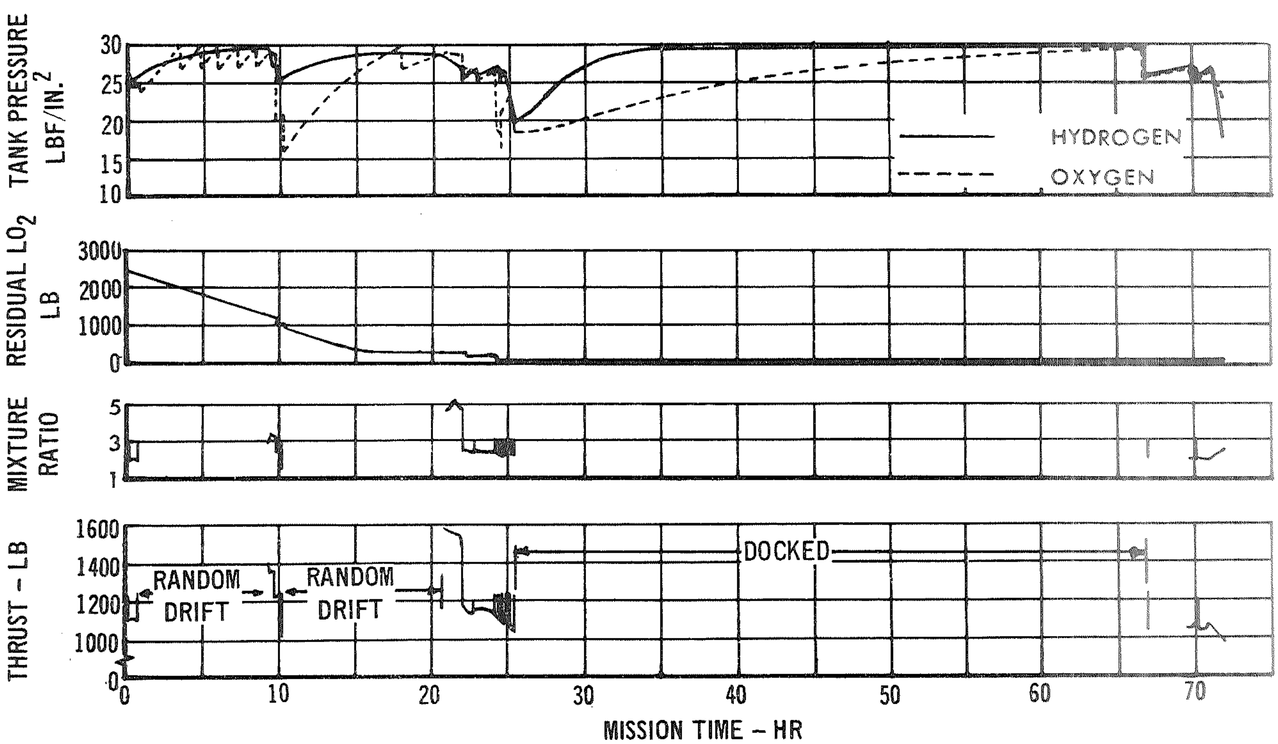
Third orbit rendezvous missions last approximately one day. Residual boiloff propellant is used for nearly the entire mission, and tank pressures and temperatures remain fairly high. The seventeenth orbit rendezvous mission is more critical, since tank pressures and temperatures drop significantly, especially during braking maneuvers just prior to docking. Also, less main engine tank liquid residuals are available for this mission. Thus, the seventeenth orbit mission establishes APS propellant requirements.

5.3 Transient Performance - The objective of this study was to investigate possible dynamic instabilities within the subsystem. APS engines can be fired continuously or in a pulse mode, with valves controlling propellant flow located adjacent to the engine. Gaseous propellant is carried from the main engine tank to valves by long lengths of tubing; therefore, rapid opening or closing of propellant valves can excite oscillations which could alter engine operation. Since fuel and oxidizer acoustic velocities are much different, and line length to the engine varies between propellants, a different oscillation frequency for each propellant could cause transient variations in engine mixture ratio. The study was confined to feed and engine assemblies, because the large main engine tank volume effec-



ORBITER MISSION DUTY CYCLE – 3RD ORBIT RENDEZVOUS

FIGURE 5–19



ORBITER MISSION DUTY CYCLE – 17TH ORBIT RENDEZVOUS

FIGURE 5–20

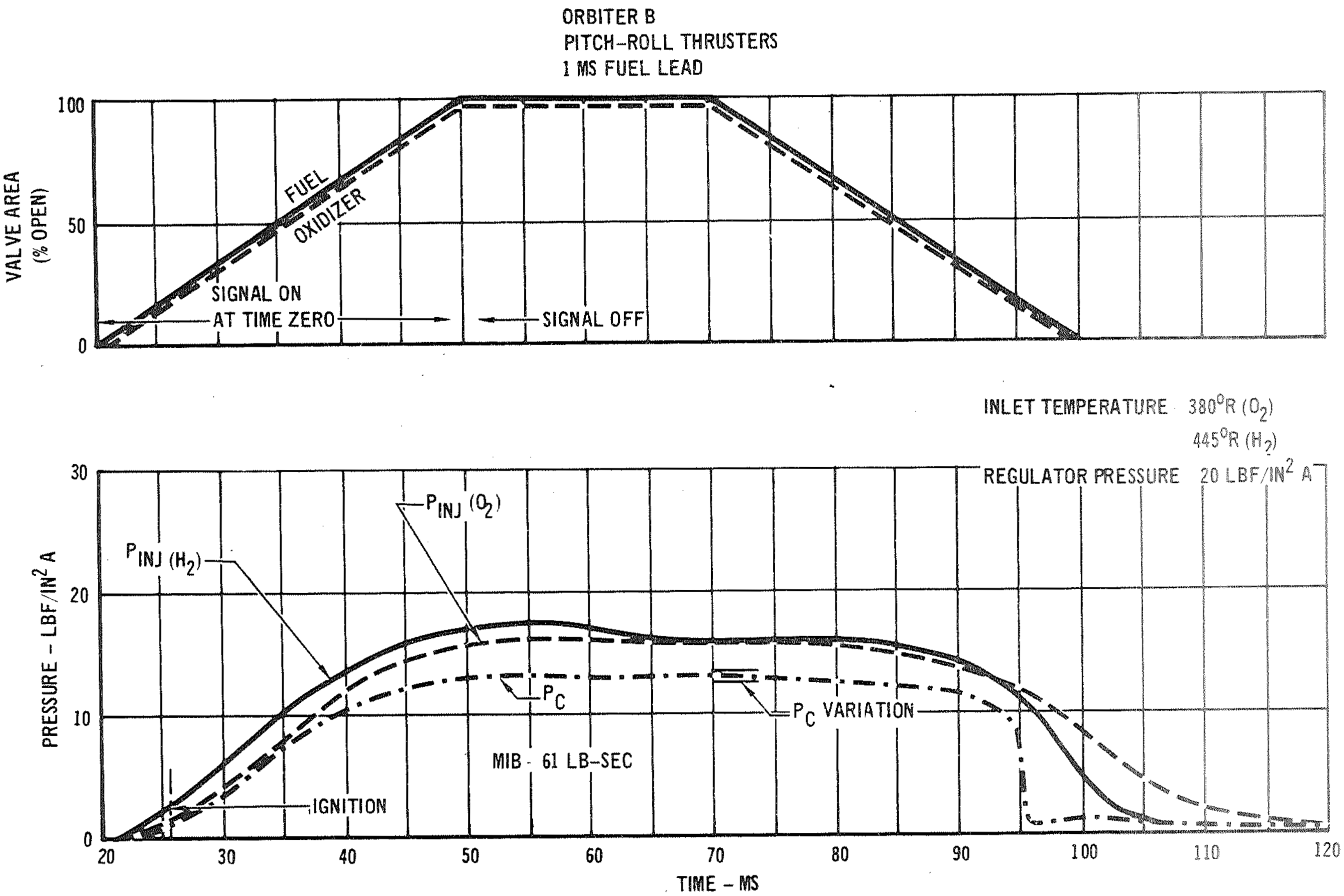
tively decouples these assemblies from the remainder of the subsystem at that point.

To determine engine-feed assembly transient characteristics, a digital computer program was developed to simulate engine start and shutdown transients. Transient models were developed for lines, valves, orifices, regulators, and engines, and were integrated into a common computer program to model the distribution network downstream of the main engine tank. Program output includes a time history of temperature, pressure, and weight flow at any desired location. In addition, engine performance parameters such as specific impulse, mixture ratio, and chamber gas temperature are calculated.

Typical engine start and shutdown characteristics are shown in Figure 5-21. Data are for a single pitch-roll engine, operating in a limit cycle mode with relatively warm propellants (445°R hydrogen, 380°R oxygen). Ignition occurs 26 ms after the valve on signal is received, and 90 percent of steady state chamber pressure is achieved at 42 ms. Propellant valve operation was delayed 20 ms to allow for pneumatic pilot valve response, and the off signal was given when valves reached full open (50 ms). The resulting impulse bit was 61 lb-sec, and steady state chamber pressure was 13 lbf/in^2a. Effect of inlet temperatures on impulse bit are shown in Figure 5-22.

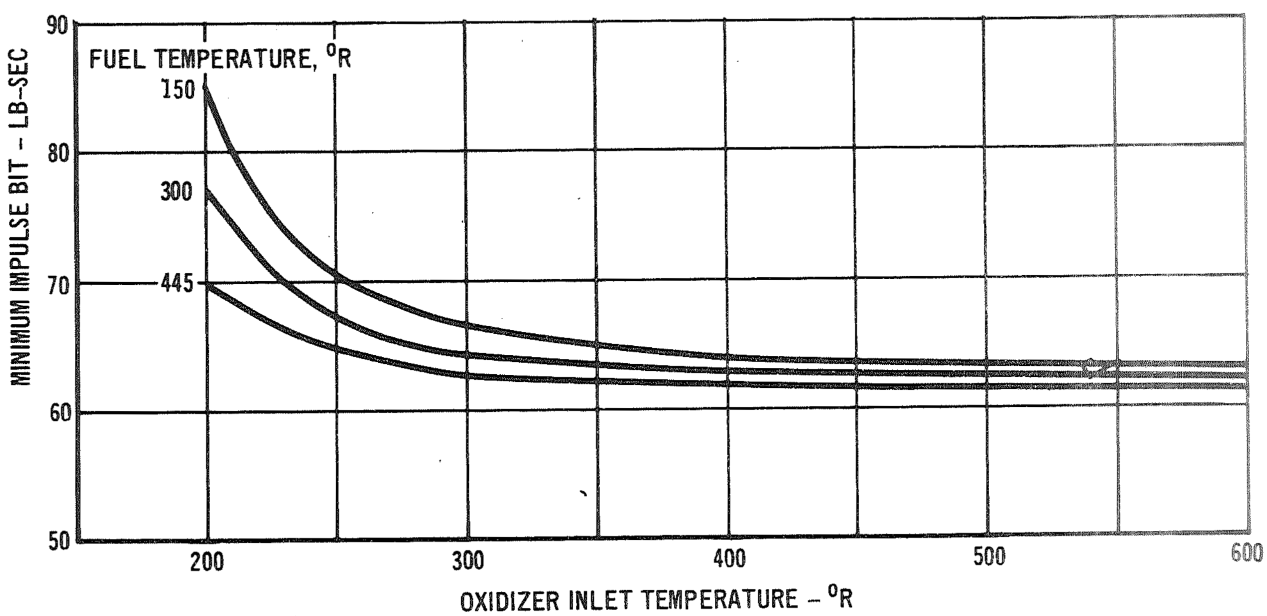
Control engine response characteristics with +X maneuver engines firing and cold propellants (150°R hydrogen, 200°R oxygen) are shown in Figure 5-23. Although +X engine firings contribute to increased line flow, the low temperatures obtained from liquid vapor mixing actually result in higher control engine chamber pressure than is realized during limit cycle operation.

Line surge pressures were evaluated using worst case conditions of six +X translation engines firing and maximum inlet pressure of 30 lbf/in a, corresponding to a failed-open regulator and maximum main tank vent pressure. Surge pressures of 42 and 23 percent were obtained in hydrogen and oxygen lines, respectively. Effect of line diameter on surge pressures is shown in Figure 5-24. Increases in line diameter have relatively little effect on surge pressures.



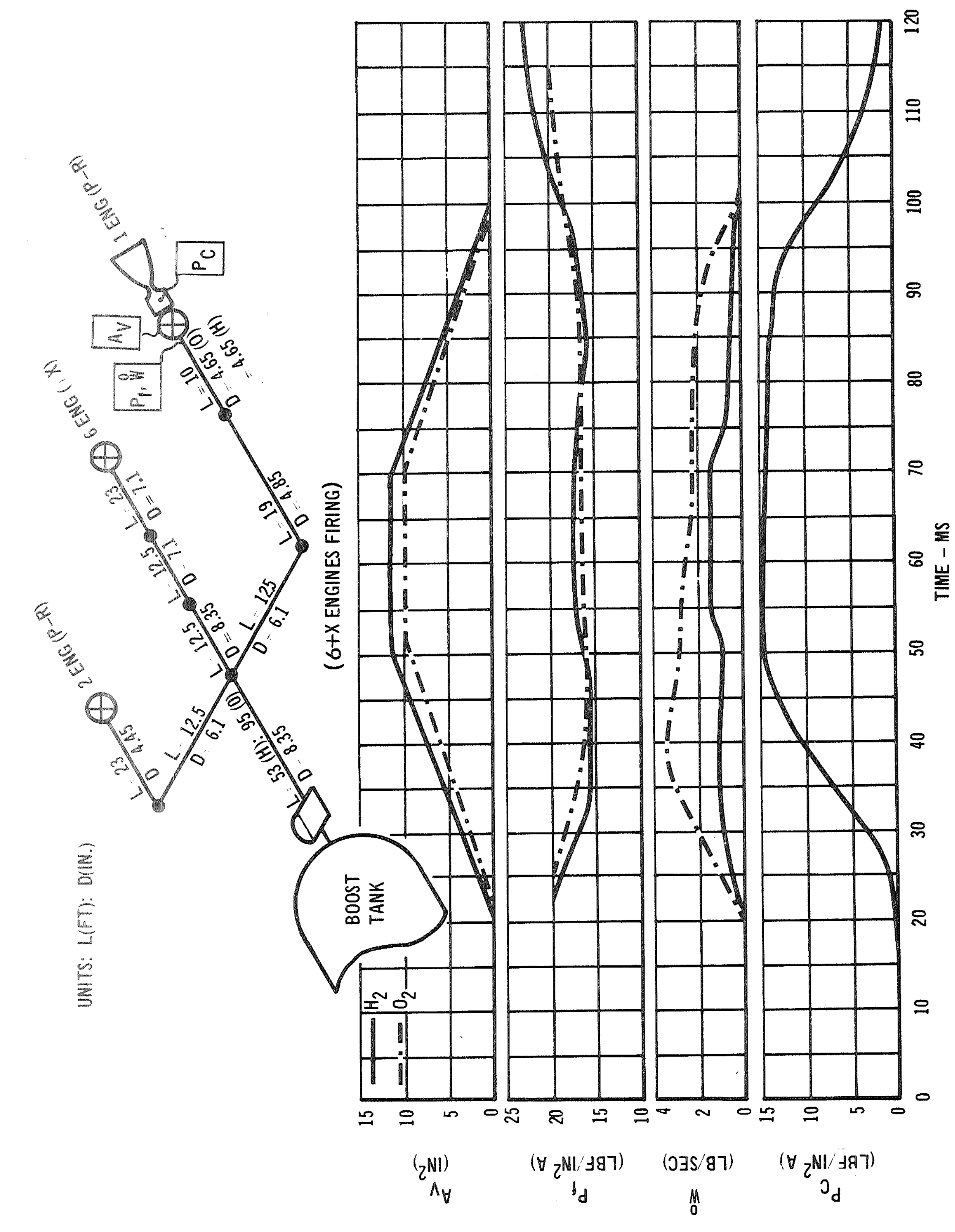
ENGINE RESPONSE CHARACTERISTICS

FIGURE 5-21



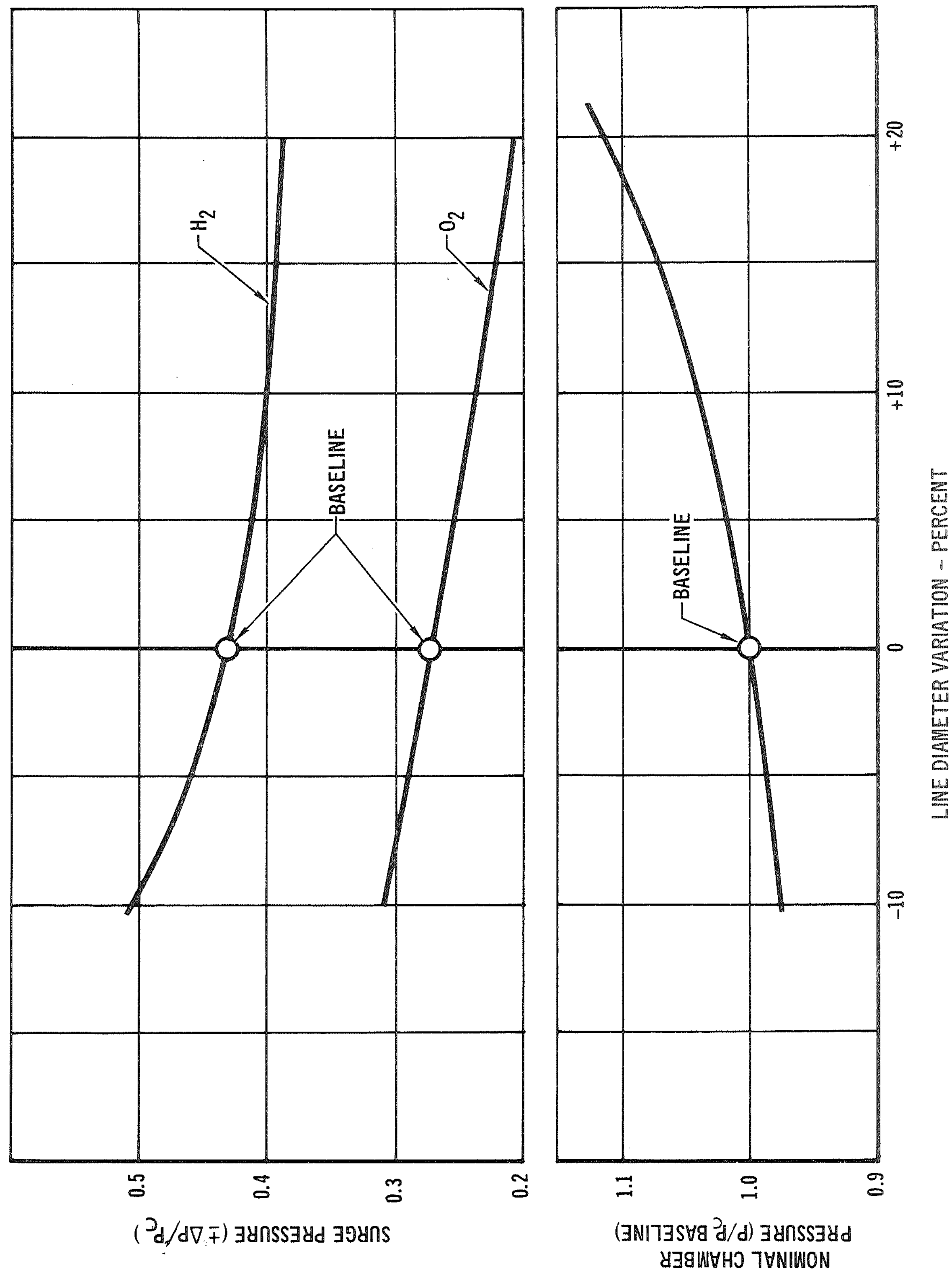
ORBITER B
SENSITIVITY OF MIB TO INLET TEMPERATURE - ORBITER APS ENGINES

FIGURE 5-22



PITCH ENGINE TRANSIENT PERFORMANCE

FIGURE 5-23



- ORBITER B
- +X ENGINE FIRINGS, SIMULTANEOUS SHUTDOWN
- INLET PRESSURE – 30 LBF/IN^2 A

ENGINE LINE SURGE PRESSURES

FIGURE 5-24

6. COMPONENT/ASSEMBLY DESIGN AND INSTALLATION

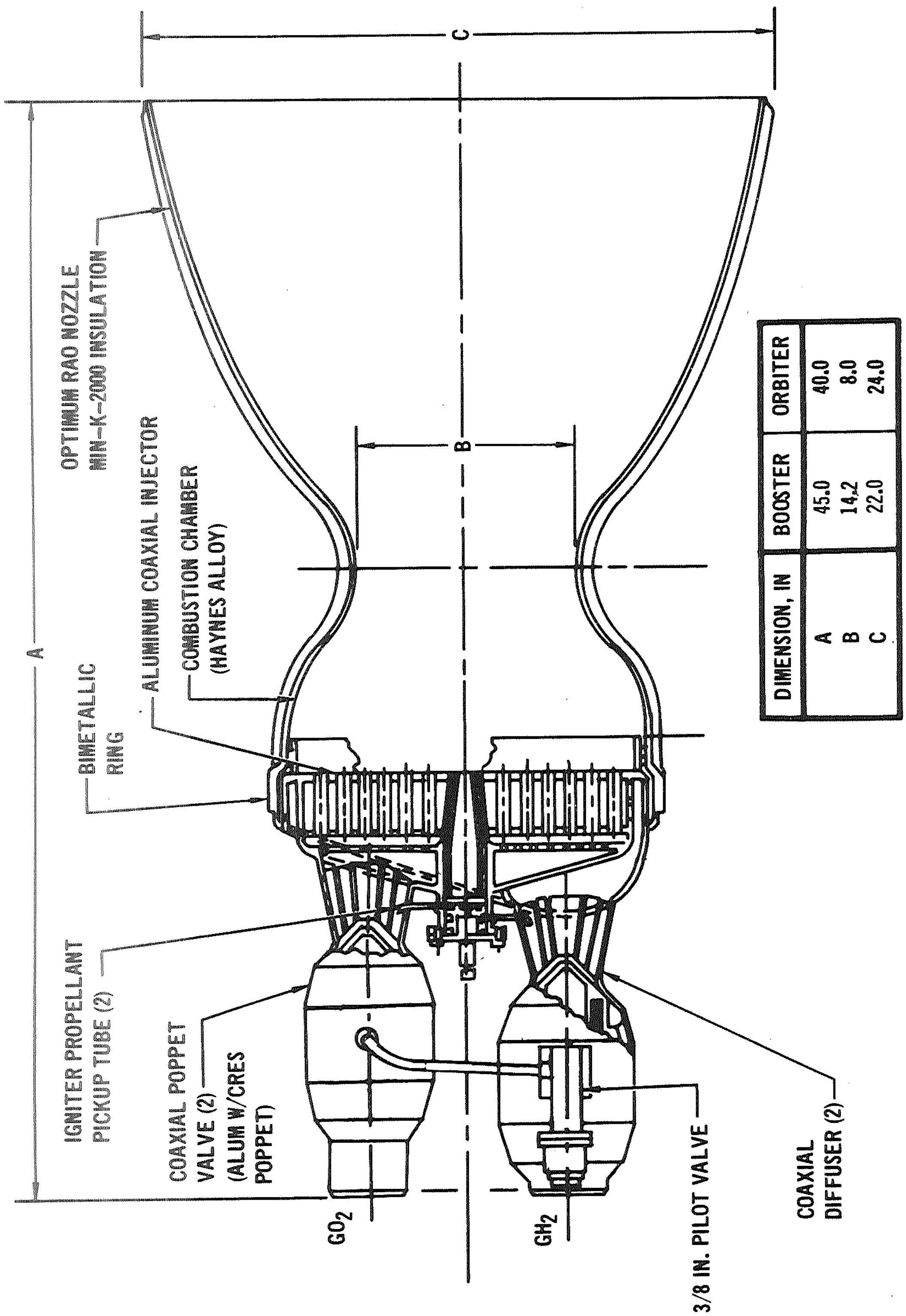
Component trade studies, component design selection and definition, and component placement and installation are discussed in this section. Performance characteristics of the most critical components (thruster, heat exchangers and liquid/vapor mixers) are discussed in Section 5. Design characteristics for the major components and subassemblies are described below. Characteristics peculiar to boosters or orbiters are noted where applicable.

6.1 Engines - Engine assemblies include propellant control valves, injector, combustion chamber, and nozzle. Orbiter engine design features and dimensions are shown in Figure 6-1. Engine cooling is achieved by hydrogen film cooling along the interior of the combustion chamber and nozzle wall. Combustion chambers and nozzles are fabricated of thin wall, high temperature Haynes Alloy while the head end is fabricated of aluminum to minimize assembly weight. The two subassemblies are attached by welding to a bimetallic ring. Chambers and nozzles are externally insulated with Min-K-2000 insulation. Ignition is achieved by a sequenced electric spark torch, while fast response, minimum pressure drop, coaxial poppet valves provide good pulse performance.

The booster engine is similar to the orbiter engines. Differences result from higher thrust level and lower expansion ratio. Booster engines are designed for an expansion ratio of 2:1; a conical nozzle shape was selected to minimize divergence losses. The orbiter engine uses an expansion ratio of 8:1 and has a Rao nozzle contour. A summary of engine physical characteristics is shown in Figure 6-2 and engine weight characteristics are shown in Figure 6-3.

6.2 Valve and Actuation System - Candidate low pressure valves are shown in Figure 6-4. The type of valve employed depends on the particular valve application. Coaxial poppet valves were selected for the engines to achieve requisite cycle life, sealing, and response characteristics. Motor operated, visor-type valves shown in Figure 6-5 were selected for isolation valves where high cycle life and fast response are not required. Motor operated iris valves (Figure 6-6) were the choice for pressure regulators since they provide low pressure drop at full flow conditions and exhibit a linear stroke flowrate relationship (linear K_W curve).

Figure 6-7 shows the engine valve in the closed (non-actuated) position. A single pilot valve controls both H_2 and O_2 valves on each engine. Total valve response time including pilot valve is 50 ms, helium actuation requirements are 0.52×10^{-3} lb/cycle. A schematic of the orbiter pneumatic subsystem is shown in



DIMENSION, IN	BOOSTER	ORBITER
A	45.0	40.0
B	14.2	8.0
C	22.0	24.0

ENGINE ASSEMBLY

FIGURE 6–1

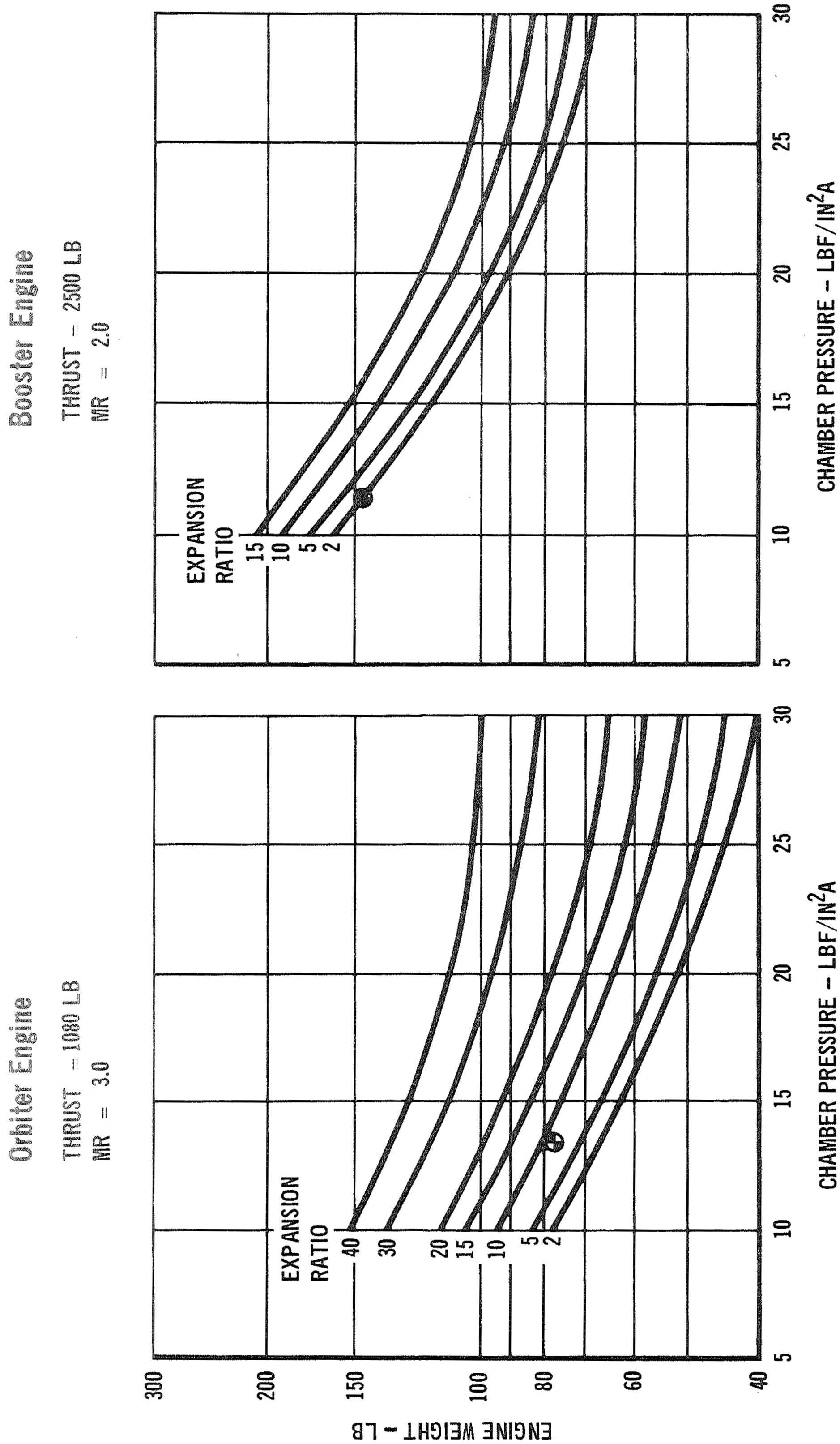
THRUST	LB	1080
CHAMBER PRESSURE	LBF/IN.^{2}A	13.7
MIXTURE RATIO		3.0
EXPANSION RATIO		8:1
INLET PRESSURE	LBF/IN.^{2}A	15.7
INLET TEMPERATURE	O_2/H_2 – ^oR	200/150
SPECIFIC IMPULSE	SECS	376.5
FLOW RATE (TOTAL	LB/SEC	2.87
FUEL FILM COOLANT	%	15
CYCLE LIFE	CYCLES	100,000
WEIGHT – TOTAL	LB	77.0
– INJECTOR		38.7
– CHAMBER & NOZZLE		14.6
– PROPELLANT VALVES		15.7
– IGNITION & MISCELLANEOUS		8.0
DIMENSIONS	– IN.	
– OVERALL LENGTH		40.0
–THROAT DIAMETER		8.0
– INSIDE CHAMBER DIAMETER		13.0
– NOZZLE EXIT DIAMETER (O.D.)		24.0
– INTERFACE DIAMETER		19.0
– VALVE EQUIVALENT FLOW AREA – IN.^{2} (O_2/H_2)		4.15/4.90

ORBITER APS ENGINE DESIGN CHARACTERISTICS

THRUST	LB	2500
CHAMBER PRESSURE	LBF/IN.^{2}A	11.0
MIXTURE RATIO		2.0
EXPANSION RATIO		2:1
INLET PRESSURE	LBF/IN.^{2}A	14.0
INLET TEMPERATURE	O_2/H_2 – ^oR	400/150
SPECIFIC IMPULSE	SECS	342
FLOW RATE (TOTAL	LB/SEC	7.31
FUEL FILM COOLANT	%	10
CYCLE LIFE	CYCLES	100,000
WEIGHT – TOTAL	LB	149.0
– INJECTOR		83.1
– CHAMBER & NOZZLE		22.3
– PROPELLANT VALVES		34.6
– IGNITION & MISCELLANEOUS		9.0
DIMENSIONS	– IN.	
– OVERALL LENGTH		45.0
–THROAT DIAMETER		14.2
– INSIDE CHAMBER DIAMETER		23.2
– NOZZLE EXIT DIAMETER (O.D.)		22.0
– INTERFACE DIAMETER		34.0
– VALVE EQUIVALENT FLOW AREA – IN.^{2} (O_2/H_2)		13.2/16.6

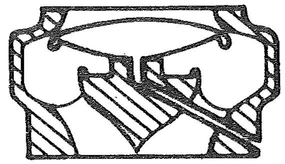
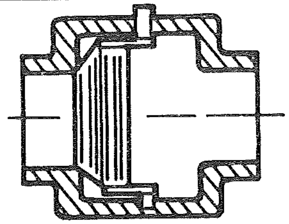
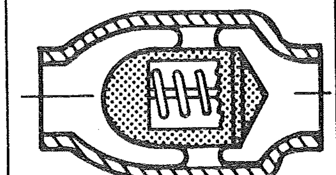
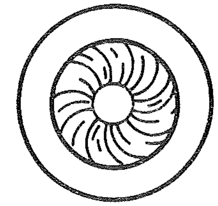
BOOSTER APS ENGINE DESIGN CONDITIONS

FIGURE 6–2



LOW PRESSURE FILM-COOLED ENGINE WEIGHTS
Includes Valves

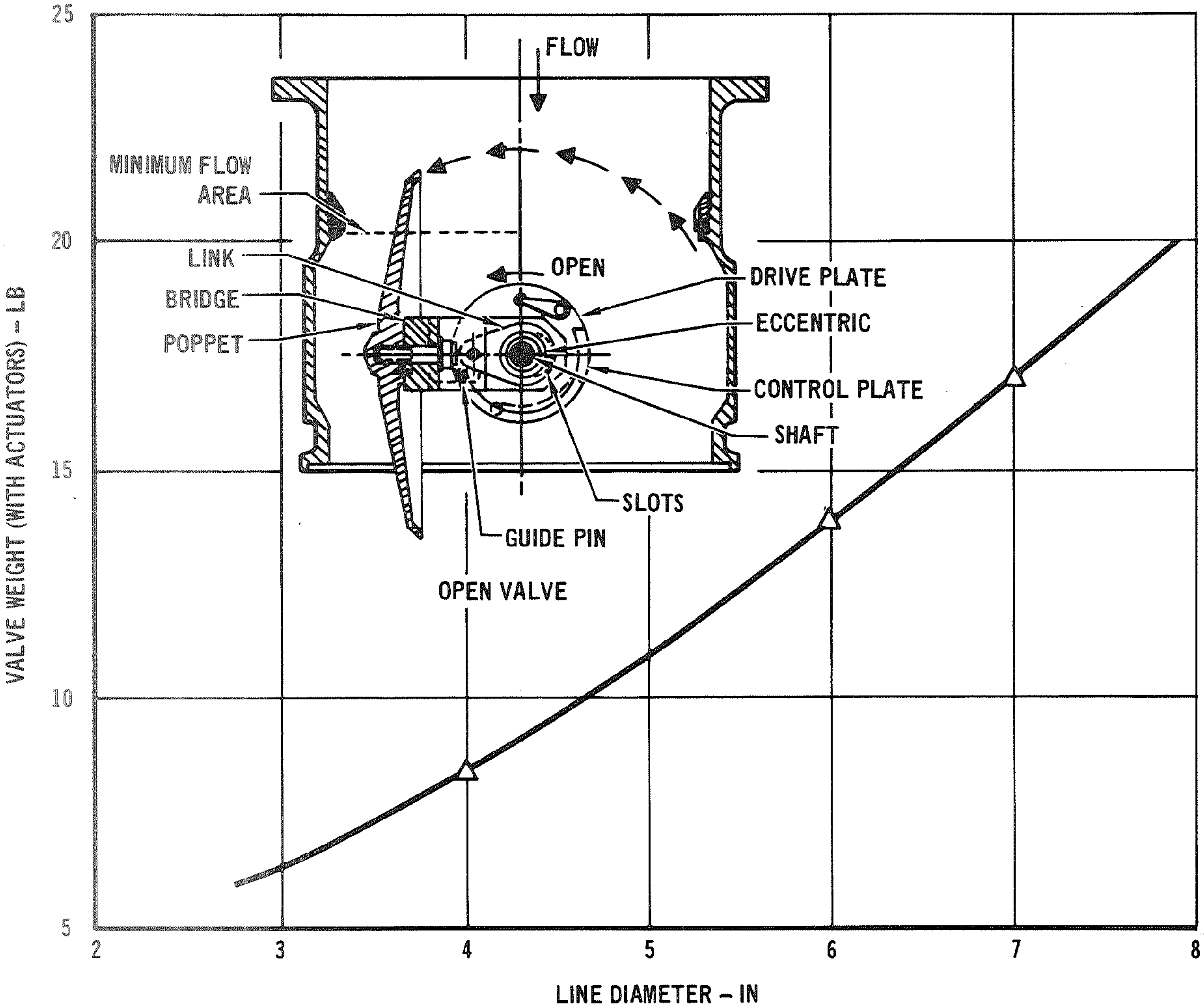
FIGURE 6-3

APPLICATION/CRITERIA	VALVE CONCEPT			
	SPRING DISC DIAPHRAGM	BUTTERFLY/VISOR	COAXIAL POPPET	IRIS
				
ENGINE VALVES LOW ΔP FAST RESPONSE CYCLE LIFE OF 100,000 POSITIVE SEALING MINIMUM WT & ENVELOPE HIGH RELIABILITY	HIGH PNEUMATIC ACTUATION FLOW REQUIREMENT LIFE LIMITED	LIMITED CYCLE LIFE	(√) ATTAINABLE CYCLE LIFE; POSITIVE SEALING; 1 PSI ΔP; HIGH RELIABILITY	POSITIVE SEALING NOT POSSIBLE
ISOLATION VALVES LOW ΔP SLOW RESPONSE CYCLE LIFE OF 1000 POSITIVE SEALING MINIMUM WEIGHT AND ENVELOPE	HIGH PNEUMATIC ACTUATION FLOW REQUIREMENT	(√) MINIMUM ENVELOPE; ATTAINABLE CYCLE LIFE; LOW DP; LIGHT WEIGHT MORE COMPLEX MECHANISM FOR VISOR	ACCEPTABLE ALTERNATE (HEAVIER)	POSITIVE SEALING NOT POSSIBLE
PRESSURE REGULATOR FAST RESPONSE TO DEMAND HIGH RELIABILITY MINIMUM PRESSURE ERROR BAND MINIMUM WEIGHT AND ENVELOPE	NO INTERMEDIATE FLOW SETTING (HIGH CYCLE REQUIREMENTS FOR FLOW MODULATION)	LIMITED RESPONSE DUE TO DRIVE MECHANISM	NO INTERMEDIATE FLOW SETTING (HIGH CYCLE REQUIREMENTS FOR FLOW MODULATION)	(√) BROAD FLOWRATE RANGE, MINIMUM WEIGHT AND ENVELOPE; LINEAR STROKE – AREA RELATIONSHIP

(√) SELECTION

LOW PRESSURE CONTROLS SELECTION

FIGURE 6–4

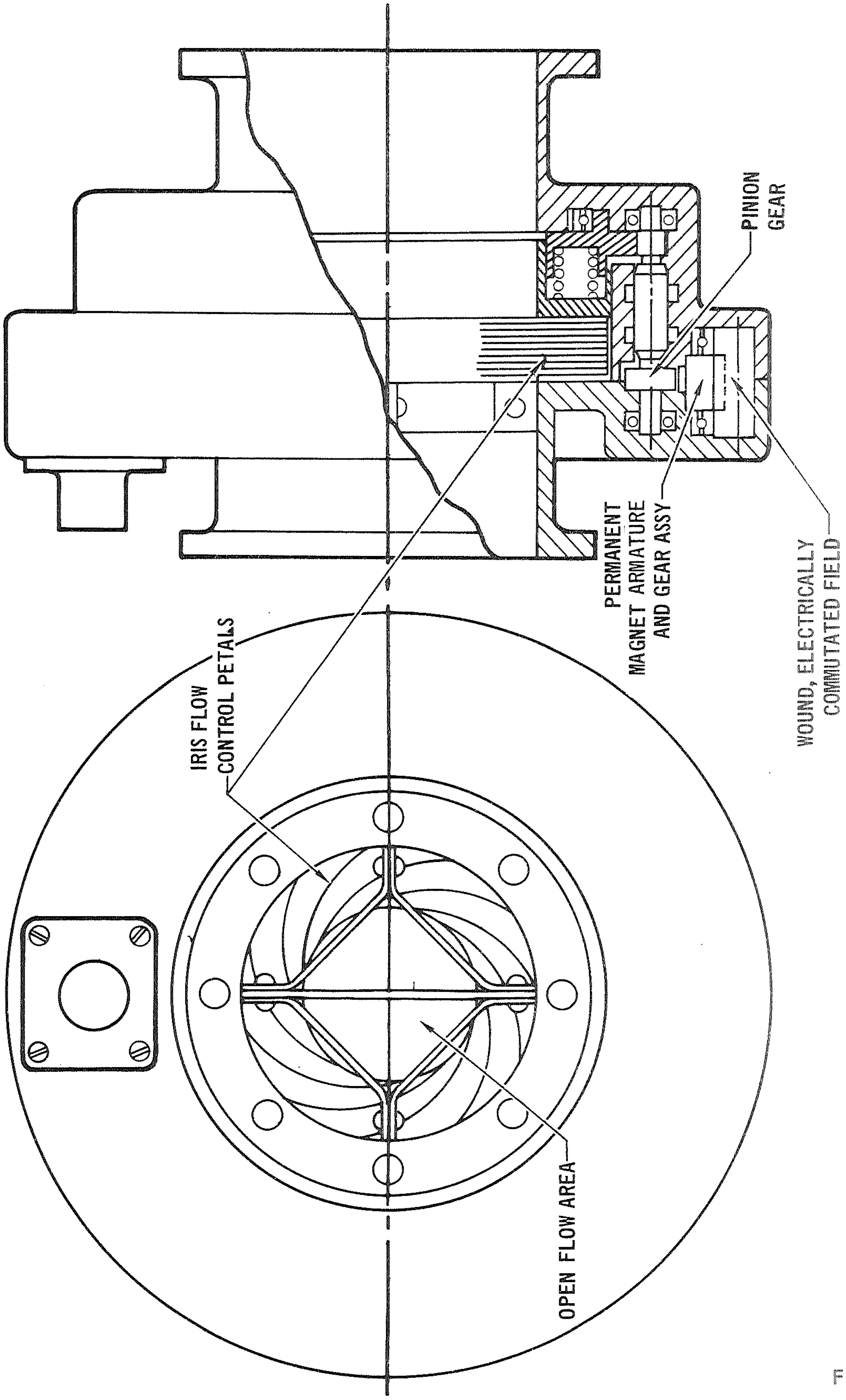


ISOLATION VALVE DESIGN AND WEIGHT
(Visor–Type)

FIGURE 6–5

Figure 6-8. The booster APS employs a similar subsystem. Three and one-half pounds of helium on the booster, and eleven pounds on the orbiter, have been provided to satisfy mission requirements. Each pneumatic circuit is backed up by a quantity measuring fuse, which shuts in the event of a broken line or failed-open pilot valve. Thus, after a single failure, the entire pneumatic supply will not be vented. The aft orbiter helium supply was separated into two storage tanks to provide fail-safe capability in case one tank is depleted. A third tank is located forward to avoid long distribution lines.

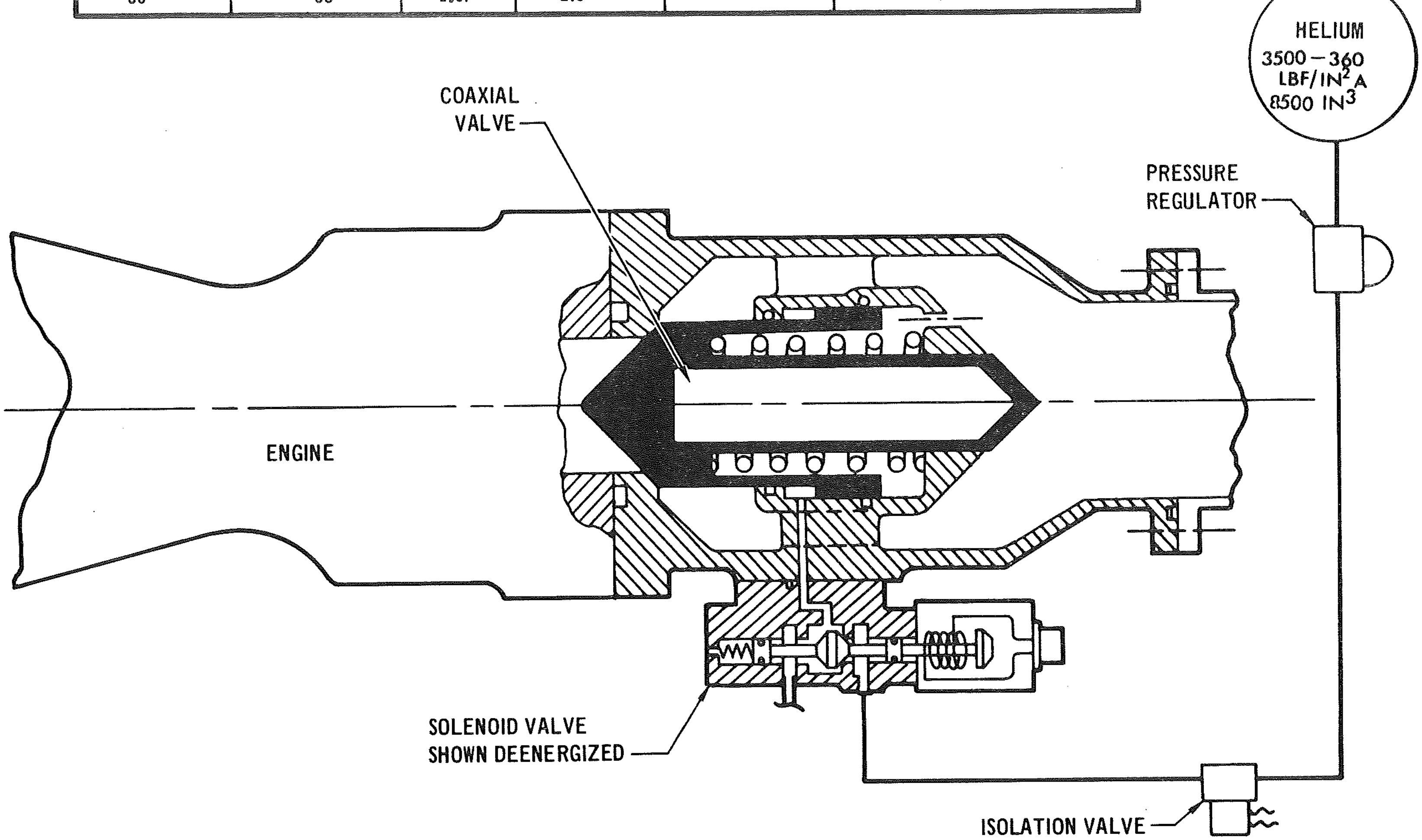
6.3 Propellant Distribution Assembly - The propellant distribution assembly supplies propellant to the engine assemblies and provides isolation of engines in



IRIS REGULATOR DESIGN

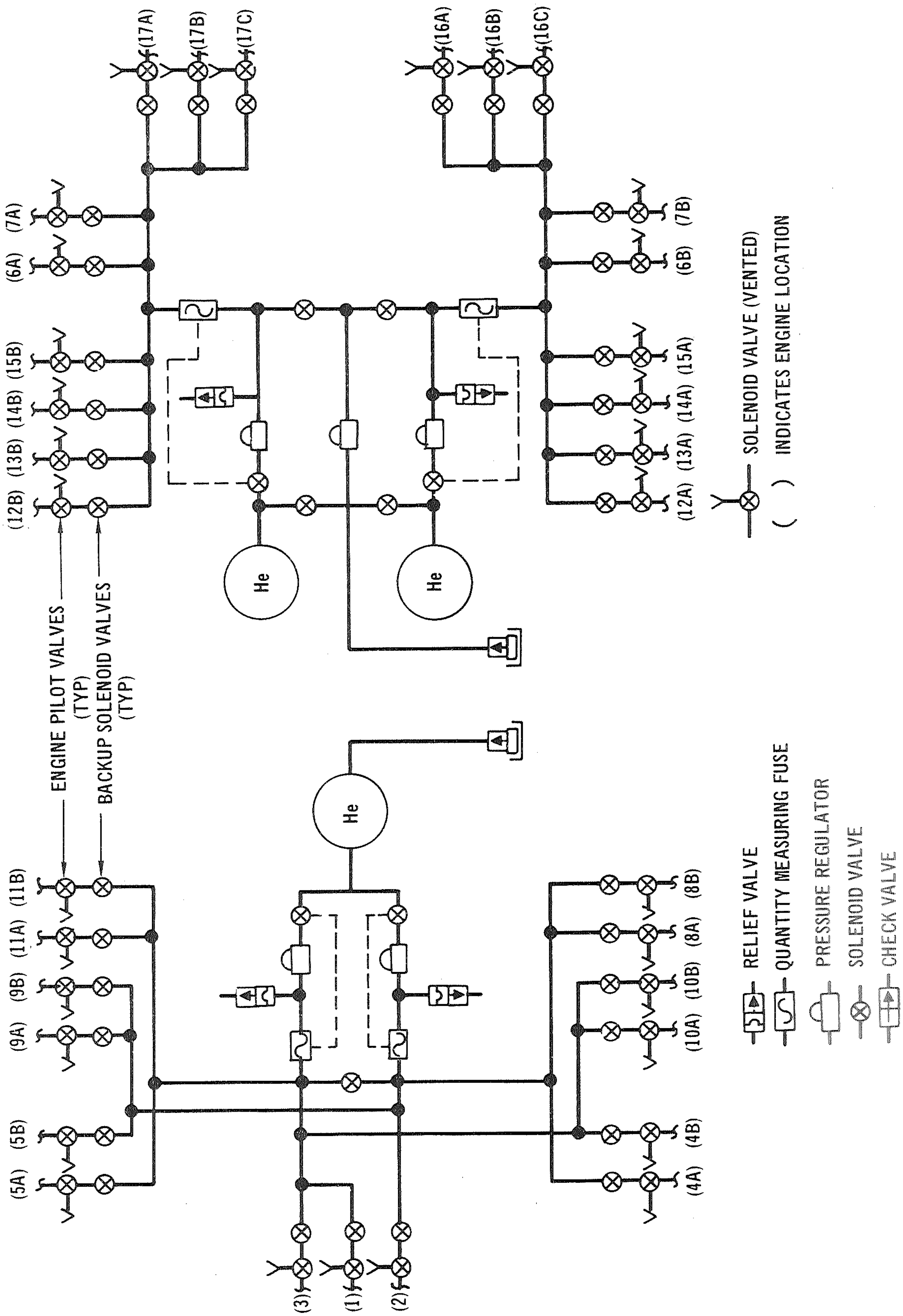
FIGURE 6–6

VALVE DESIGN				ACTUATOR REQUIREMENTS	
RESPONSE (MSEC)	SEAT LOADS (LB/IN)	STROKE (IN)	CAVITY VOL (IN^3)	PRESSURE (LB/IN^2 A)	HELIUM (LB)
50	38	1.07	2.5	250	$0.52x10^{-3}$/ENGINE CYCLE



PNEUMATIC VALVE ACTUATOR SYSTEM

FIGURE 6-7



PNEUMATIC SYSTEM – ENGINE VALVE ACTUATION – ORBITER

FIGURE 6-8

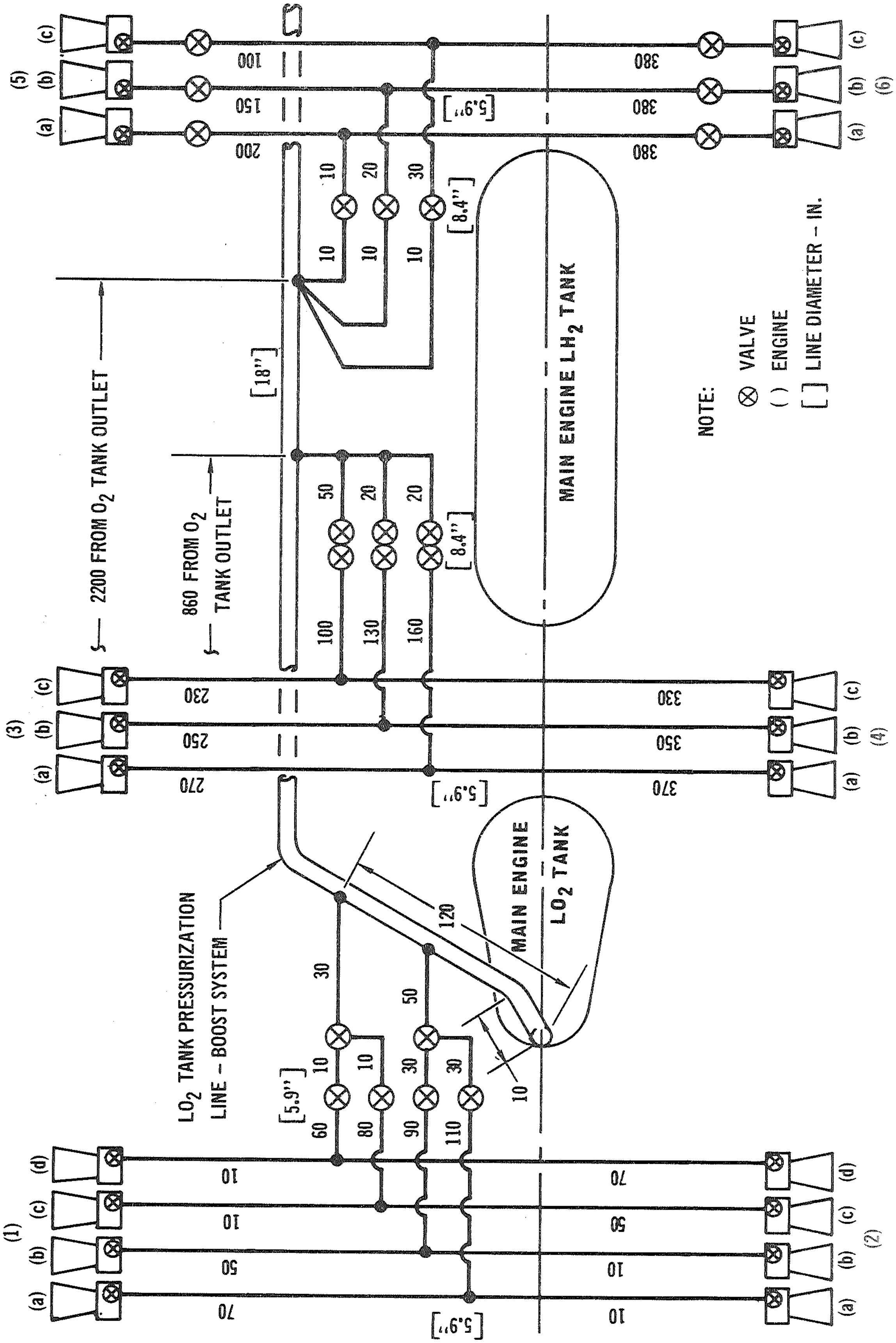
case of failure. The distribution assembly consists of ducts, isolation valves, and linear and angular compensators. Figure 6-9 presents the APS booster distribution assembly installation and Figure 6-10 shows the orbiter installation. Ducting design characteristics are summarized in Figure 6-11. The minimum gage dimensions were specified from a survey of F-4 and DC-10 aircraft ducting as shown in Figure 6-12.

Propellant ducts are fabricated of minimum gage aluminum, and main engine tank pressurization lines are used as main trunks of the APS distribution network as discussed in Appendix E. Isolation valves and manifolds were located close to the engine groups, to minimize line length and to provide isolation for each engine ring in the event of engine or valve failure. This avoided control axis cross-coupling that would otherwise be encountered when an engine failed. Each line section includes linear and angular compensators, as required, to absorb normal vehicle manufacturing tolerances, to absorb differences in thermal expansion between ducts and vehicle structure, and to compensate for structural motion. A typical line and compensator installation is shown in Figure 6-13. Aluminum bellows, used in linear and angular compensators, are significantly lighter than comparable stainless steel compensators. Although aluminum has not been extensively used for such an application in the past, data from bellows manufacturers show that aluminum bellows will be satisfactory as long as high pressures and large deflections are avoided. Detailed distribution assembly analysis and component design characteristics are presented in Appendix E.

6.4 Main Tank Liquid/Vapor Separation - Both booster and orbiter use liquid/vapor separators to prevent liquid ingestion into the distribution assembly. Due to differing requirements, the design approach between booster and orbiter varies.

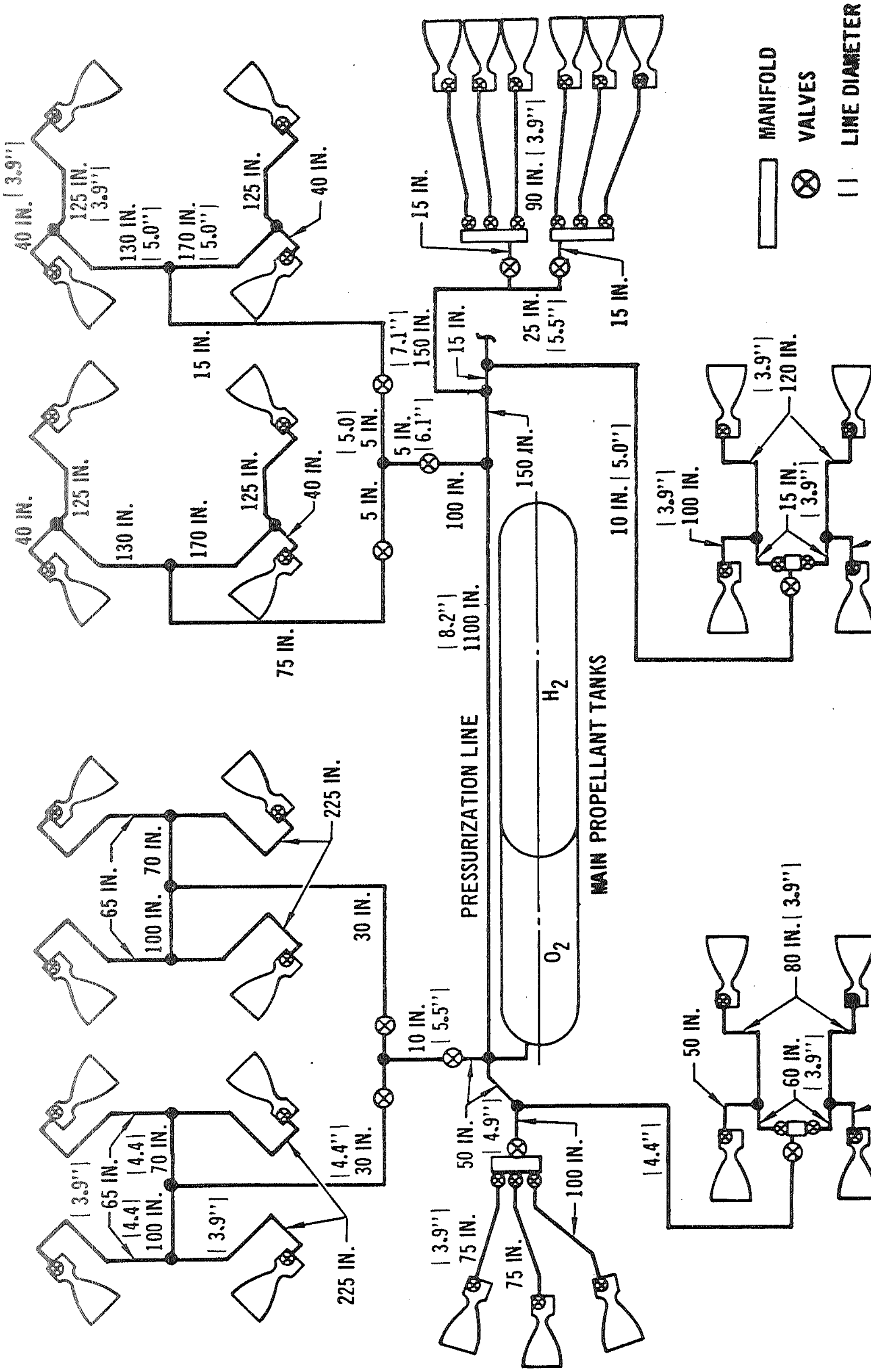
Sufficient residual propellant (liquid and vapor) remain in the booster main engine tanks to provide the entire booster APS requirements. Bulk liquid propellant is prevented from entering the APS distribution network through the use of g-sensitive valves of the type shown in Figure 6-14. Short booster mission times preclude propellant orientation in accordance with zero-g conditions, and propellant will move in relation to acceleration or inertial forces. Any shift of acceleration forces that will cause bulk liquid to move toward the valve will also close the valve from that direction. The valve is located on a standoff, ensuring that the valve will not be submerged when residuals are located along the tank wall.

The Orbiter APS is characterized by long orbital periods, where zero "g" conditions are encountered. Also, the orbiter main propulsion is shut down by



BOOSTER DISTRIBUTION NETWORK
(Oxygen Side)

FIGURE 6-9



DISTRIBUTION NETWORK
ORBITER DISTRIBUTION (OXYGEN SIDE)

FIGURE 6-10

DISTRIBUTION LINES	
MATERIAL	2219 ALUMINUM
DENSITY (LB/IN.^{3})	0.101
DESIGN TEMPERATURE (^oR)	530
ULTIMATE STRESS LBF/IN.^{2}	64,000
ULTIMATE SAFETY FACTOR	2.0
MINIMUM GAGE (INS)	
LINE DIAMETER 2–4	0.022
4–6	0.035
6–9	0.049
COMPENSATORS	
TYPE, ANGULAR	SOCKET/BELLOWS
LINEAR	IN LINE BELLOWS
MATERIAL	2219 ALUMINUM
ISOLATION VALVES	
TYPE	VISOR
MATERIAL	ALUMINUM
ACTUATION	DC REVERSIBLE MOTOR DRIVE WITH CLUTCH BRAKE

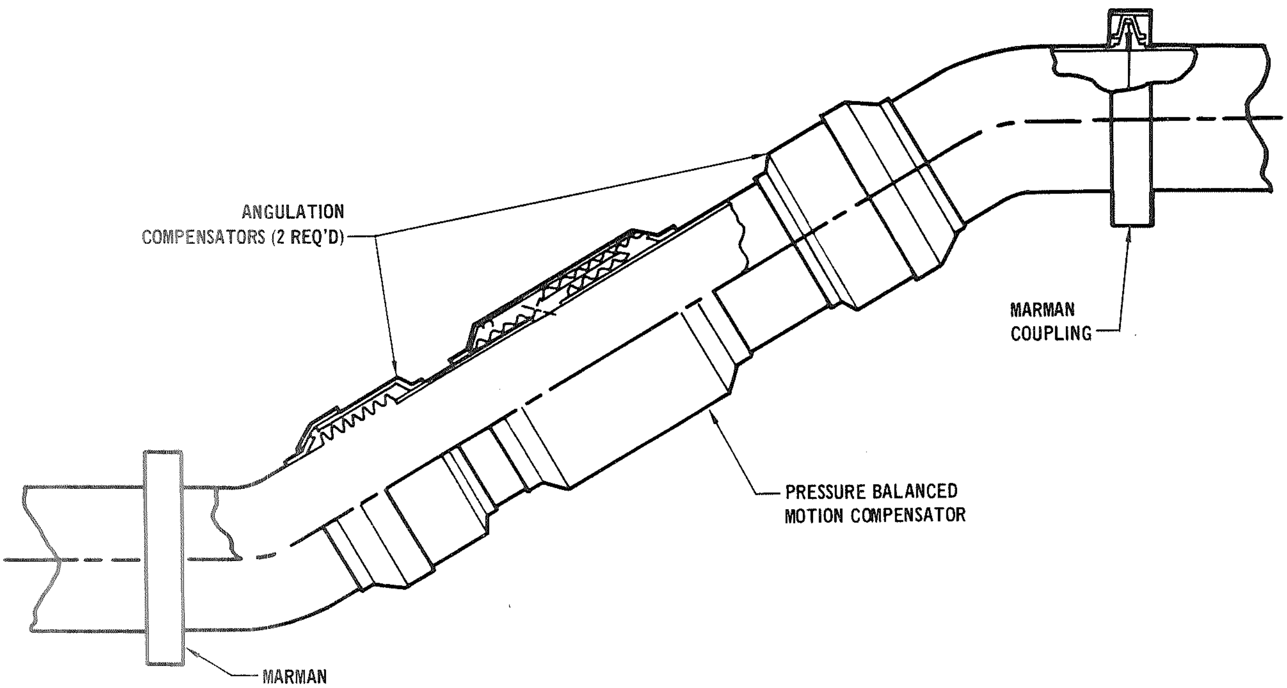
DISTRIBUTION ASSEMBLY DESIGN CHARACTERISTICS
(Oxygen and Hydrogen)

FIGURE 6–11

MODEL	USAGE	MAXIMUM DIAMETER (INCHES)	MINIMUM GAGE (INCHES)	MATERIAL	REMARKS
F–4	ENGINE BLEED & VENT	2.0 4.5	0.016 0.020	STAINLESS STEEL	0.020 MINIMUM HANDLING
F–4	FUEL	3.0	0.035	STAINLESS STEEL	
F–4	AIR CONDITIONING	2.5	0.028	ALUMINUM	MINIMUM BASED ON BENDING, LINE PRESSURE = 25 PSI
DC–10	ANTI ICE	2.5	0.025 0.035	STAINLESS STEEL	
DC–10	PNEUMATIC BLEED, ENVIRONMENTAL CONTROL	6.0	0.025	TITANIUM	TITANIUM USED TO SATISFY 400 TO 700^{o}F THERMAL ENVIRONMENT
DC–10	FUEL VENTS	3.5 4.5	0.028 0.035	ALUMINUM	0.035 MINIMUM GAGE FOR WELDING

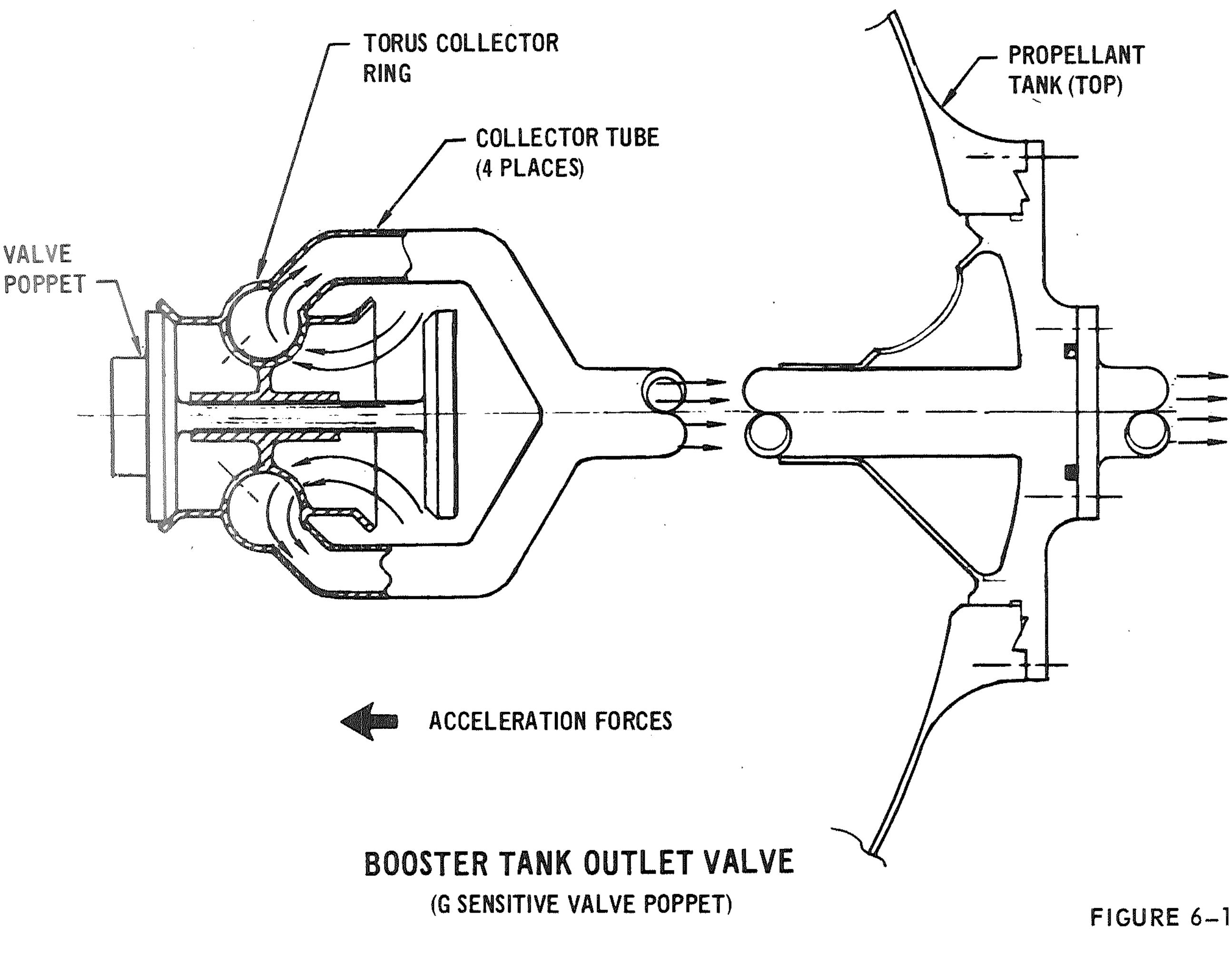
AIRCRAFT DUCTING SURVEY

FIGURE 6–12



TYPICAL LINE INSTALLATION

FIGURE 6-13



BOOSTER TANK OUTLET VALVE
(G SENSITIVE VALVE POPPET)

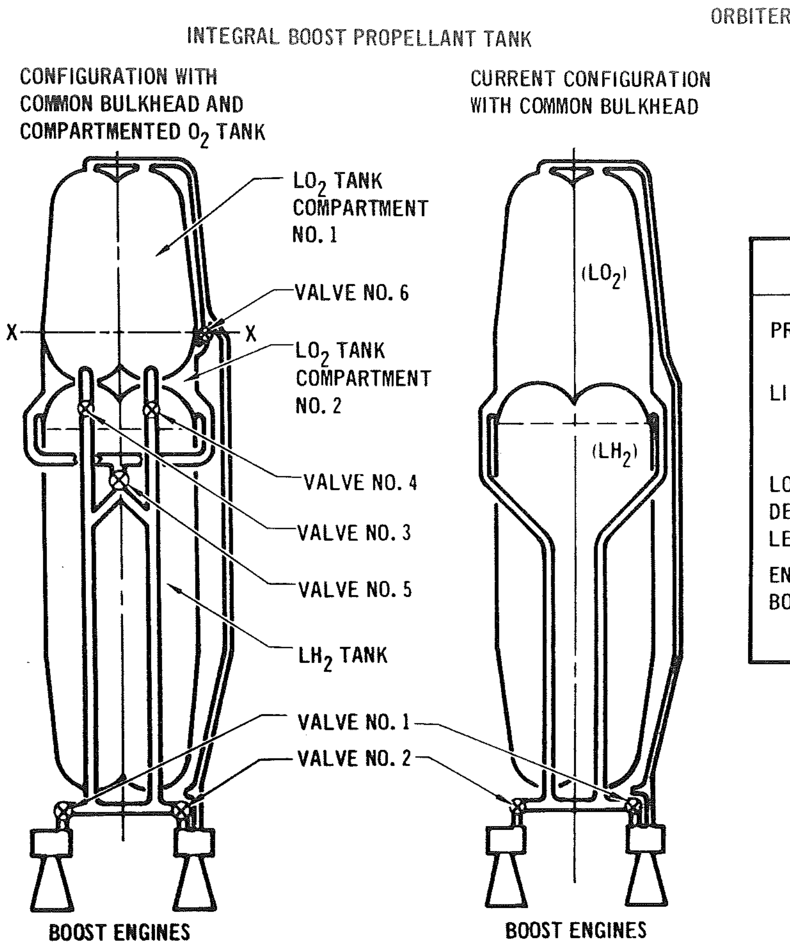
FIGURE 6-14

velocity command instead of propellant depletion; hence, a large quantity of residuals could remain at time of orbit insertion. Active liquid/vapor separators for the orbiter tank are impractical, since there is no ready means of predicting where the bulk liquid might settle in a zero g environment. Hydrogen residuals rapidly boil off and provide little, if any, APS weight advantage. For this reason, hydrogen residuals are dumped through the main engines after main engine shutdown to avoid liquid ingestion into the feed system and/or main engine tank pressure collapse. The reverse is true with liquid oxygen residuals. They boil off more slowly and represent a large potential weight savings.

A tension bulkhead, just forward of the common bulkhead, has been added in the LO_2 boost tank to contain liquid oxygen residuals. The tension bulkhead is so placed as to ensure that all LO_2 will have completely drained from the boost tank main compartment prior to main engine cutoff. The operation of this compartment tank is shown in Figure 6-15. During launch, propellant is extracted from the main tank until the liquid level reaches the top of the segmented compartment. Then, the valve to the segmented compartment is opened, allowing both tanks to drain. At shutdown, residual propellant will be located in feed lines and segmented compartment only. Although residual quantity will depend on amount of flight performance reserves utilized and specific mission requirements, at least 2440 lb will be available.

The original O_2 Tank bulkhead was designed to withstand compressive forces resulting from liquid oxygen head during launch. The added tension bulkhead will relieve the lower bulkhead of liquid oxygen head, thereby allowing a reduction in lower bulkhead weight. A summary of resultant weight is shown in Figure 6-16. The segmented compartment was obtained for a net weight penalty of only 50 lbs, including valves. This weight penalty is minor compared to potential oxygen residual availability.

6.5 Liquid/Vapor Mixer (Orbiter Only) - For major APS operations, part of the propellant is obtained from the main engine tank, the remainder supplied as liquid from the propellant storage assembly to the liquid/vapor mixer. During low demand attitude control operations, all propellant is extracted from the main engine tank; there is no downstream injection in the liquid/vapor mixer. Liquid/vapor mixer design is shown in Figure 6-17. It consists of a liquid injection element, hyperthin vanes located normal to the gas stream, and a downstream mixing length to allow liquid vaporization. Initially, when the temperature of the gas being removed from the main engine tank is high, a large quantity of liquid can be



SEQUENCE OF OPERATION FOR CONFIGURATION WITH COMPARTMENTED O_2 TANK

EVENT	VALVE OPERATION	REMARKS
PRELAUNCH	NOS. 1 & 2 CLOSED NOS. 3, 4, 5, 6 OPEN	LOAD COMPARTMENTS NOS. 1 & 2
LIFTOFF	ALL VALVES OPEN EXCEPT NO. 5	ALL LO_2 FLOW FROM COMPARTMENT NO. 1
LO_2 SUPPLY DEPLETED TO LEVEL X–X	OPEN NO. 5	LO_2 FLOW FROM COMPARTMENTS NO. 1 & 2
END OF BOOST	CLOSE ALL VALVES EXCEPT NO. 5	ALL O_2 RESIDUALS ARE CONTAINED IN COMPARTMENT NO. 2.

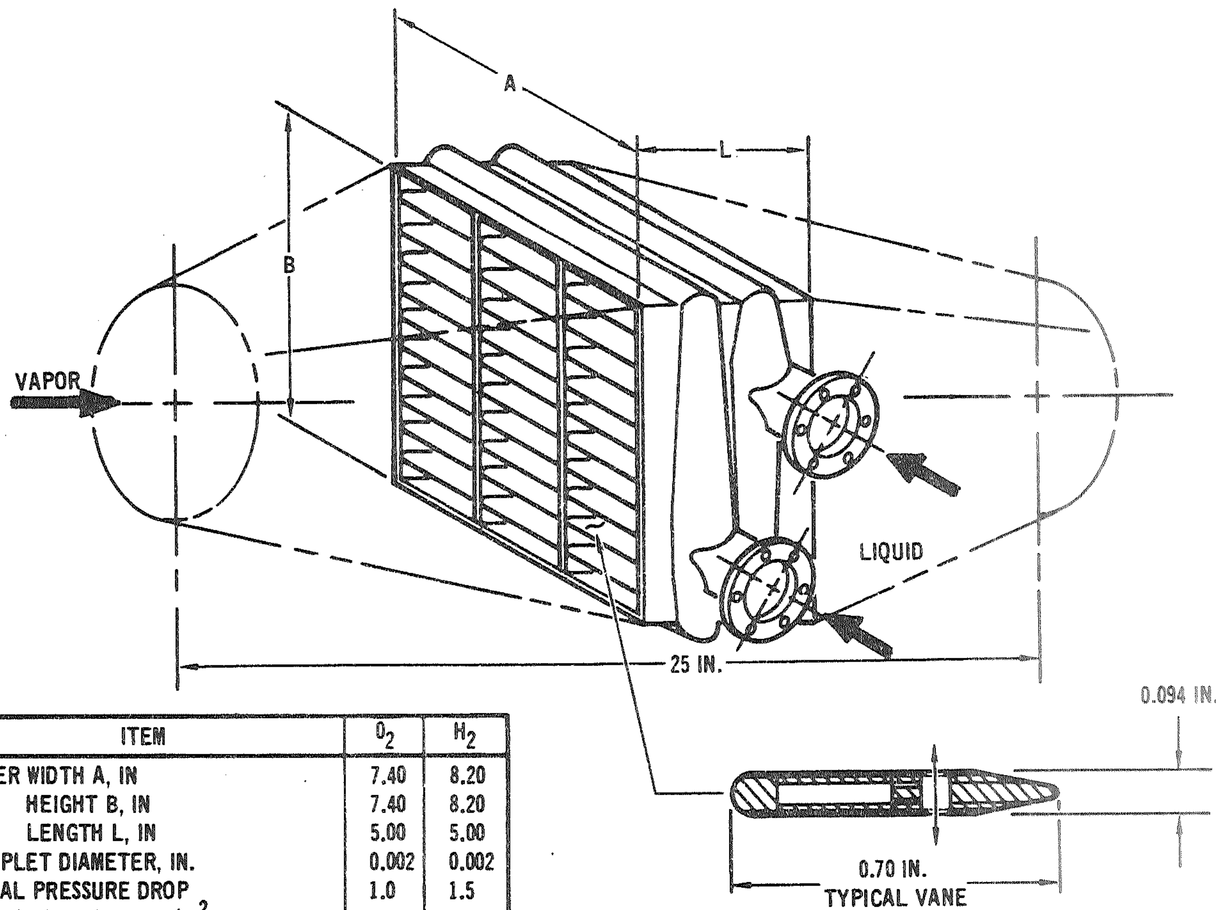
COMPARTMENTED OXYGEN PROPELLANT TANK

FIGURE 6–15

ORBITER TANK WEIGHT COMPARISON	BOOST TANK CONFIGURATION	
ITEM	COMMON BULKHEAD	COMPARTMENTED O_2 TANK
COMMON BULKHEAD	950 LB	540 LB
COMPARTMENT BULKHEAD	–	210 LB
VALVES	–	175 LB
LINES AND MOTION COMPENSATORS	–	75 LB
	950 LB	1000 LB

ORBITER TANK WEIGHT COMPARSION

FIGURE 6–16

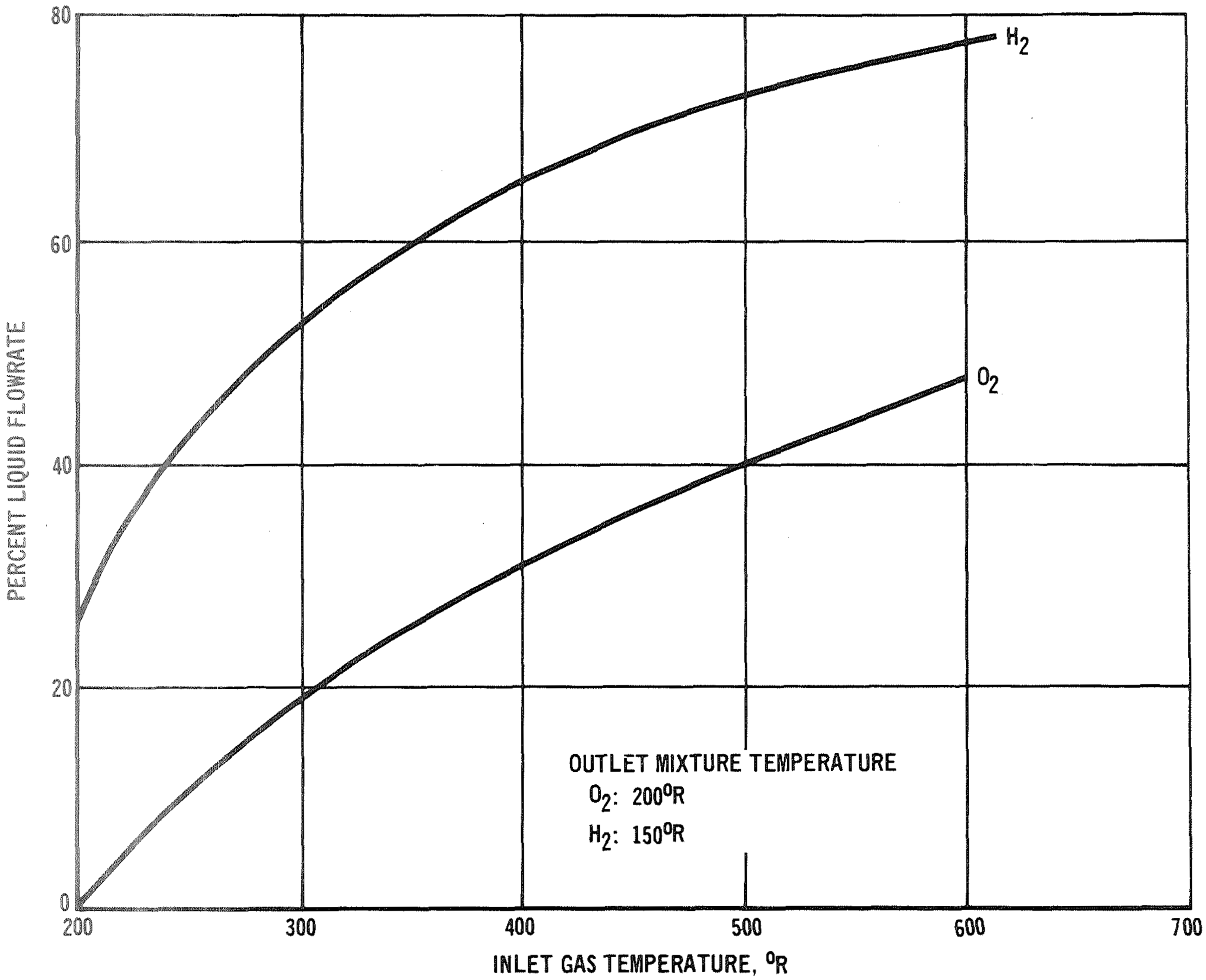


ITEM	O_2	H_2
MIXER WIDTH A, IN	7.40	8.20
HEIGHT B, IN	7.40	8.20
LENGTH L, IN	5.00	5.00
DROPLET DIAMETER, IN.	0.002	0.002
TOTAL PRESSURE DROP GAS STREAM, LBF/IN^2	1.0	1.5
PRESSURE DROP, LIQUID, LBF/IN^2	10.0	7.8
GAS STREAM MACH NO.	0.2	0.2
LIQUID DISCHARGE VELOCITY,MAX,FPS	20	47

PROPELLANT LIQUID/VAPOR MIXER

FIGURE 6–17

used. As tank temperature and pressure decay with time, liquid flow rate must be decreased to maintain desired mixer outlet conditions. More propellant must then be extracted from the main engine tank to support steady state conditions at the mixer outlet. Figure 6-18 gives the liquid flow rates required to maintain desired

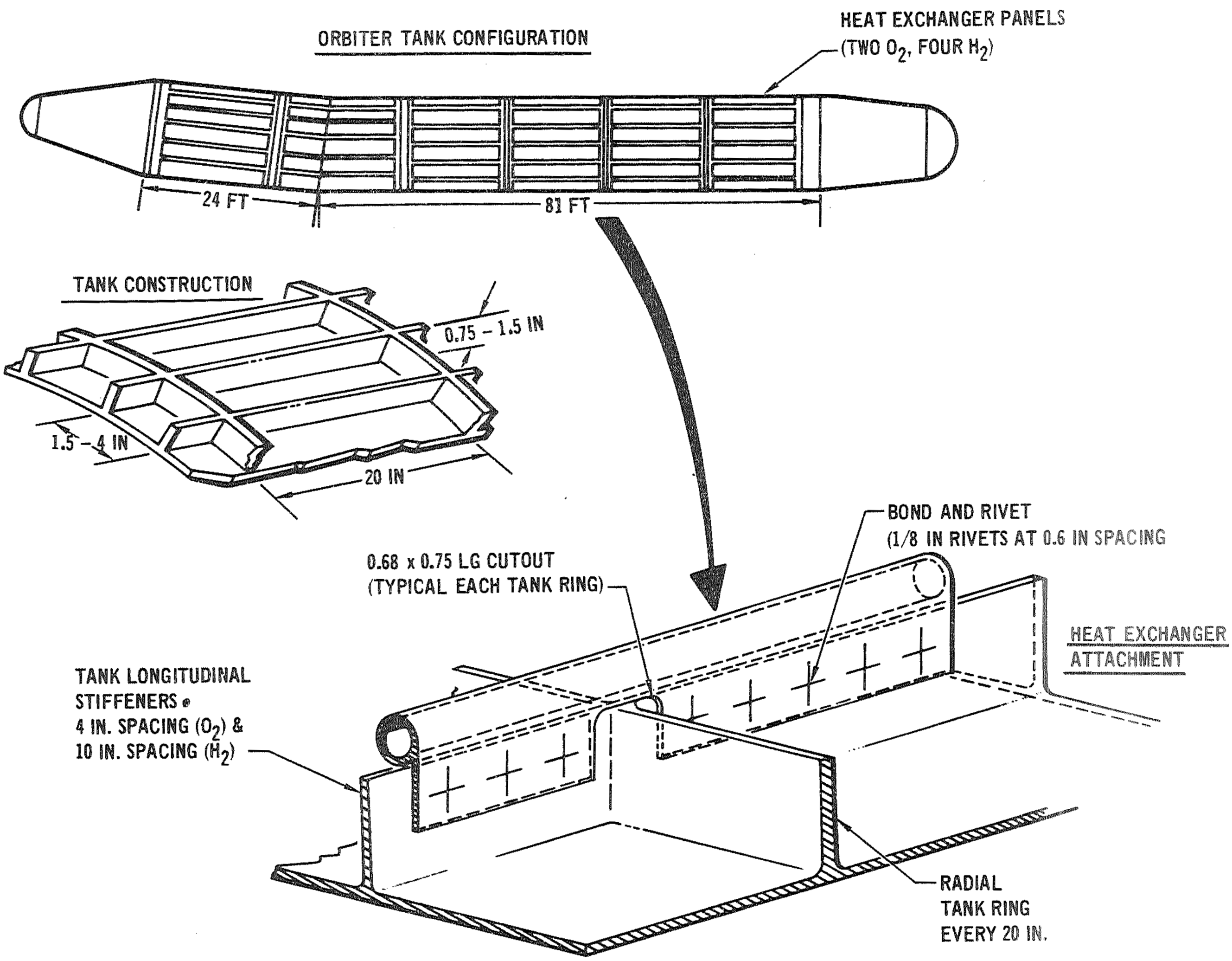


LIQUID/VAPOR MIXER
LIQUID FLOW RATE TO PROVIDE REQUIRED OUTLET TEMPERATURES

FIGURE 6-18

outlet temperatures. Flow control is provided by a motor driven cavitating venturi throttle valve which decouples the flow control from pressure oscillations in the mixer. Mixer weights are 11.4 and 17.4 lb for hydrogen and oxygen, respectively. These weights do not include approximately 11 lbs. for a liquid throttle valve.

6.6 Heat Exchanger (Orbiter Only) - Orbiter heat exchangers are used to condition resupply liquid propellant when main engine tank pressure is less than 24 lbf/in^2. Resupply propellant is vaporized, then superheated in the heat exchanger. The heat exchanger consists of thin wall, aluminum tubing mounted directly on the tank walls (which serve as a heat sink). Figure 6-19 illustrates



PASSIVE HEAT EXCHANGER CONCEPT

FIGURE 6-19

tube mounting to tank longitudinal structural stiffeners. Tube and mounting section modulus adds to longitudinal rib stiffness, permitting a reduction in tank rib height and thus weight. However, the weight reduction can only be applied to the O_2 tank since the H_2 rib height is at the minimum required for riveting.

The oxygen heat exchanger is divided into two panels, each with 154 tubes, approximately 0.4 in. in diameter. The hydrogen heat exchanger is divided into four panels, each consisting of 62 tubes, 0.3 in. in diameter. These panels are connected in parallel, and serve to reduce heat exchanger frictional pressure drop. Specific heat exchanger design characteristics were defined on the basis of subsystem performance (as presented earlier in Paragraph 4.1), and are shown in Figure 6-20.

TYPE	MULTIPLE TUBE/HEAT SINK	
LOCATION	INTEGRAL WITH MAIN ENGINE TANK WALL	
ATTACHMENT	TUBE FLANGE RIVETED TO TANK LONGITUDINAL STIFFENERS	
TUBE CHARACTERISTICS		
MATERIAL	2014-T6 ALUMINUM	
DENSITY, LBM/IN^3	0.101	
DESIGN TEMPERATURE, ^oR	530	
ULTIMATE STRESS, LBF/IN^2	64,000	
ULTIMATE SAFETY FACTOR	2.0	
MINIMUM GAGE, INCHES	0.022	
MAXIMUM MACH NUMBER	0.3	
PANEL DIMENSIONS	OXYGEN	HYDROGEN
NUMBER OF PANELS	2	4
NUMBER OF TUBES	154	62
TUBE LENGTH, FT	17.5	15.0
TUBE SPACING, IN	4.0	10.0
TUBE DIAMETER, IN	0.394	0.298

HEAT EXCHANGER DESIGN CHARACTERISTICS

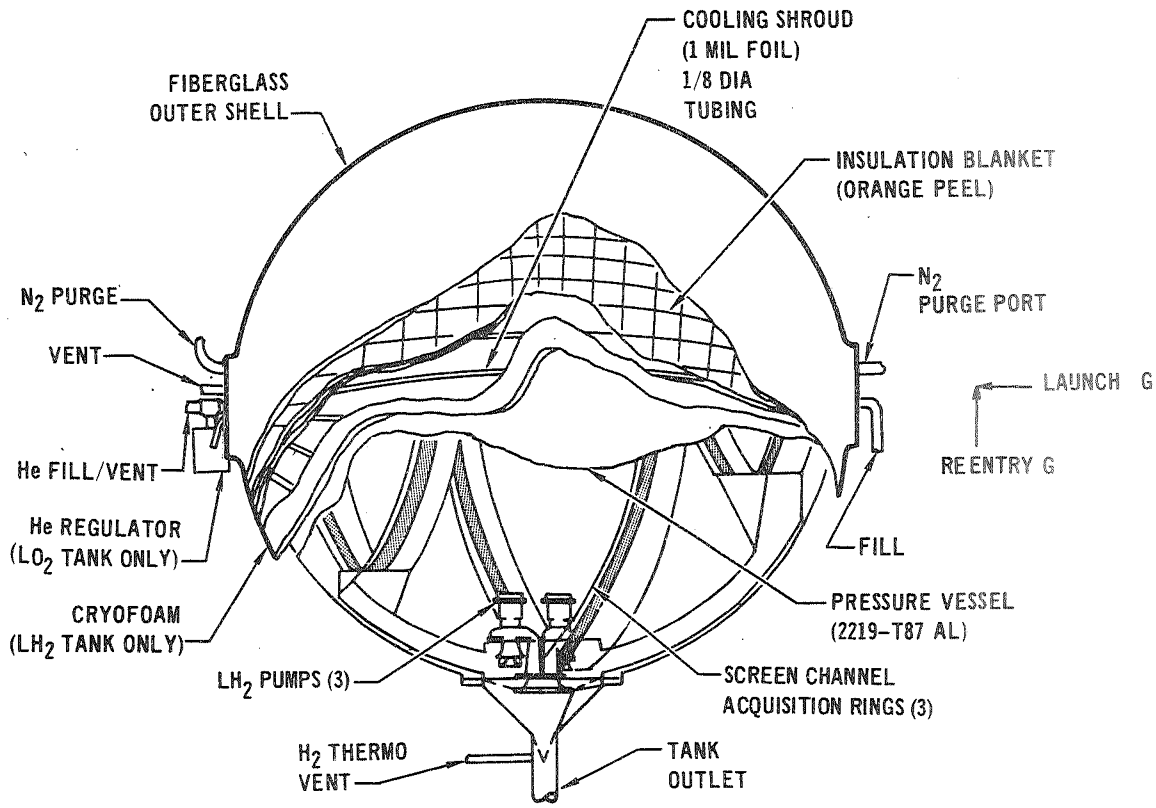
FIGURE 6-20

6.7 Propellant Storage - After boost, sufficient residual propellants are available in the booster main engine tanks to meet APS requirements. Therefore, auxiliary booster APS storage is not required. However, in the case of the orbiter impulse requirements are much greater, and auxiliary propellant storage is required. A detailed summary of the study performed to define orbiter APS tank design characteristics is presented in Appendix D, and a brief summary of selected tank design is described below.

Orbiter APS LH_2 and LO_2 are stored in spherical tanks located immediately forward of the payload bay. This location provides ready access for maintenance through the payload bay. The tanks are supported at three points by a low conductivity support structure. Tank support legs are made of fiberglass tubes, 6.0 in. in diameter and ranging in thickness from 0.035 in. to 0.10 in. Pin joints attachment points accommodate deflections caused by loads and thermal expansions.

The LO_2 tank is pressurized by regulated helium pressure. The He pressurant storage tank is mounted inside the LO_2 tank to take advantage of the volumetric efficiency gained by storing the pressurant at LO_2 temperatures. Hydrogen pressure is provided by low head rise boost pumps located in the tank sump. The LH_2 tank will be prepressurized with helium to 40 lbf/in^2a. As the LH_2 supply is depleted, tank pressure will drop to approximately 20 lbf/in^2a. This pressure will maintain an NPSP of at least 0.5 lbf/in^2a at pump inlets, and will suppress nucleate boiling (which could occur during operation at saturated conditions).

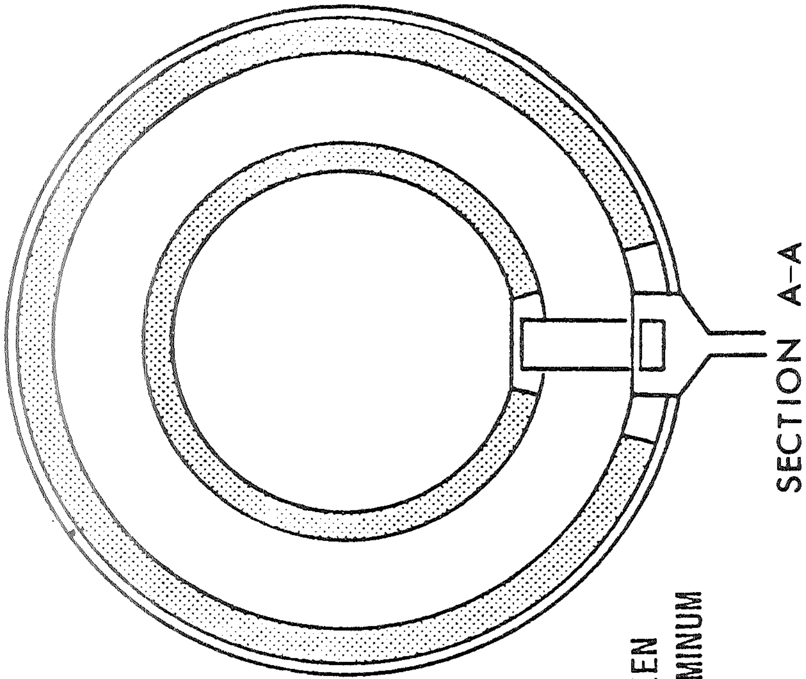
The tankage concept consists of a 2219 aluminum basic pressure vessel, a layer of cryofoam (on the LH_2 tank only), a cooling shroud made of 0.125 in. diameter aluminum tubing brazed to a very thin aluminum shroud, a blanket of high performance insulation (HPI) and a fiberglass outer shell. Figure 6-21 illustrates the



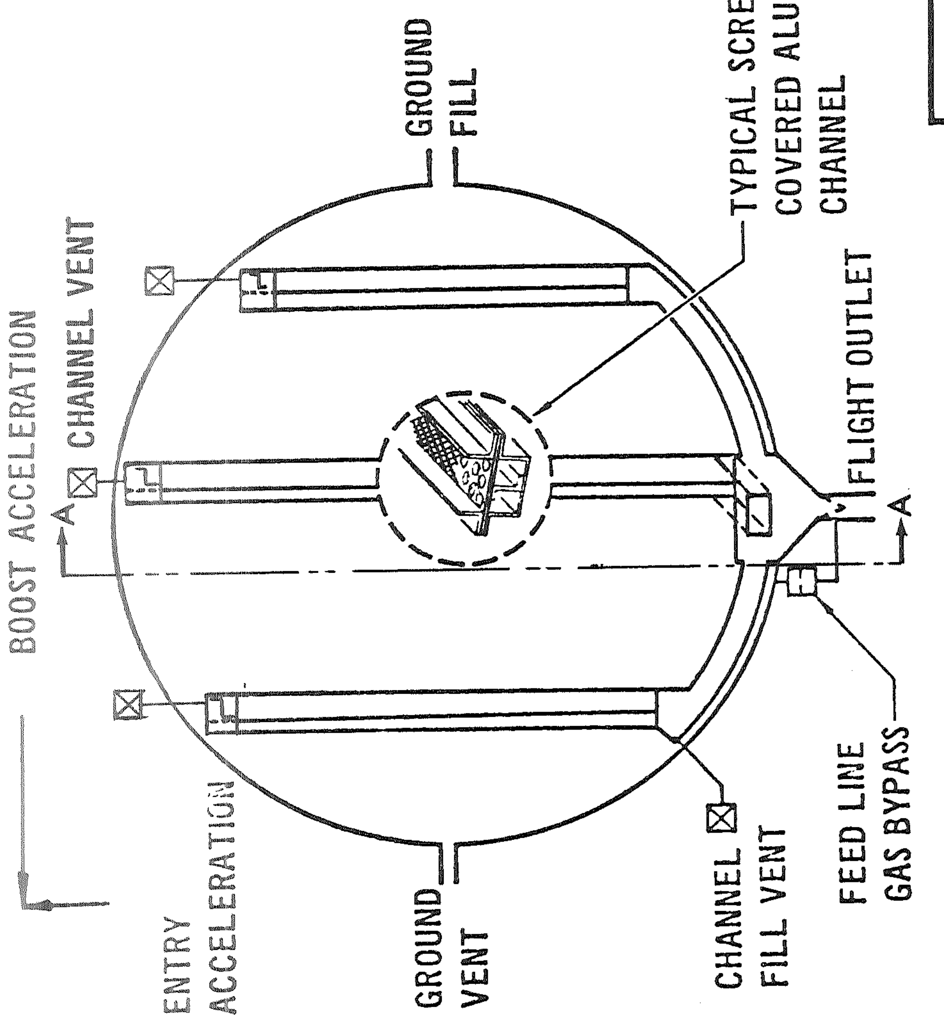
PROPELLANT TANK INSULATION/COOLING CONCEPT FIGURE 6-21

tank assembly. Cooling shroud temperature is maintained by continuous hydrogen vent. LH_2 is extracted for cooling, expanded to provide a 7°R temperature drop, then routed to shroud, tank supports, and penetrations, where vaporization is completed. The fiberglass outer shell serves as an environmental shield for the tank insulation. On the ground, constant purge of GN_2 provides an inert atmosphere which protects the HPI against contamination and corrosion. In orbit, the fiberglass shell is vented to vacuum to enable the HPI to function efficiently. On re-entry, helium (for hydrogen tank) or nitrogen (for oxygen tank) is purged through the cavity between the outer shell and the tank to prevent the shell from collapsing, and to prevent atmospheric contamination of the HPI.

Propellant acquisition is accomplished through the use of screen channels shown in Figure 6-22. Screen channels are so located that some portion of the



PROPELLANT	SCREEN SIZE	CHANNEL SIZE (IN)
O_2	250 x 1370 (17μ) 200 x 600 (40μ)	3 x 6
H_2	325 x 2300 (10μ) 200 x 600 (40μ)	5 x 8



APS PROPELLANT ACQUISITION DEVICE

FIGURE 6–22

screen will always be "wetted," thereby ensuring communication with the outlet.

A summary of the tankage assembly weights are shown in Figure 6-23.

	HYDROGEN (LB)	OXYGEN (LB)
STRUCTURAL COMPONENTS		
PROPELLANT TANK	224	51
FIBERGLASS SHELL (INCLUDING SUPPORTS)	98	22
PROPELLANT ACQUISITION DEVICE	105	37
TANK MOUNTS	62	50
TANK INSULATION		
FOAM SUBSTRATE	31	–
INSULATION BLANKET	99	32
INSULATION SUPPORTS	10	3
COOLANT LOOP		
SHROUD	28	2
VALVES, CONTROLS	4	4
PRESSURIZATION	159	32
TOTAL	820	233

PROPELLANT STORAGE ASSEMBLY – TANK WEIGHT SUMMARY

FIGURE 6-23

7. SUBSYSTEM WEIGHT AND DESIGN SENSITIVITIES

7.1 Booster APS - A weight breakdown of the booster APS is presented in Figure 7-1. No propellant is required; main engine tank residuals provide the total mission requirements. Use of main engine residuals results in a weight penalty associated with increased main engine subsystem weight. The weight penalty incurred is:

pressurant (H_2 only)	660 lb
liquid/vapor separation valves	141 lb
heat exchanger (H_2 only)	57 lb
Total weight penalty	858 lb

The 660 lb H_2 pressurant weight penalty represents the increase in vapor residual associated with the reduction in initial hydrogen tank vapor temperature as discussed in Paragraph 4.1. The total weight penalty (858 lb) has been assessed against the APS, and is included in the total booster APS weight of 5647 lb. The remainder of the subsystem consists of engine assemblies (3081 lb) and distribution assembly (1708 lb).

COMPONENT (NO.)	WEIGHT, LB O_2	WEIGHT, LB H_2
PROPELLANT	NONE REQUIRED, MAIN ENGINE TANK RESIDUALS ARE UTILIZED	
MAIN ENGINE PROPULSION MODS	(63)	(795)
PRESSURANT PENALTY	0	660
LIQUID/VAPOR SEPARATION VALVES (1)	63	78
HEAT EXCHANGER	0	57
PROPELLANT DISTRIBUTION ASSEMBLY	(657)	(1051)
LINES	227	350
COMPENSATORS, LINEAR (23)	104	251
COMPENSATORS, ANGULAR (46)	64	105
ISOLATION VALVES (21)	262	345
ENGINE ASSEMBLIES	(3081)	
ENGINES (20)	2980	
PNEUMATIC SUBASSEMBLY		
HELIUM	3.5	
TANKS (3)	34.0	
VALVES (28)	12.5	
REGULATORS (3)	9.0	
LINES	42	
TOTAL	(5647)	

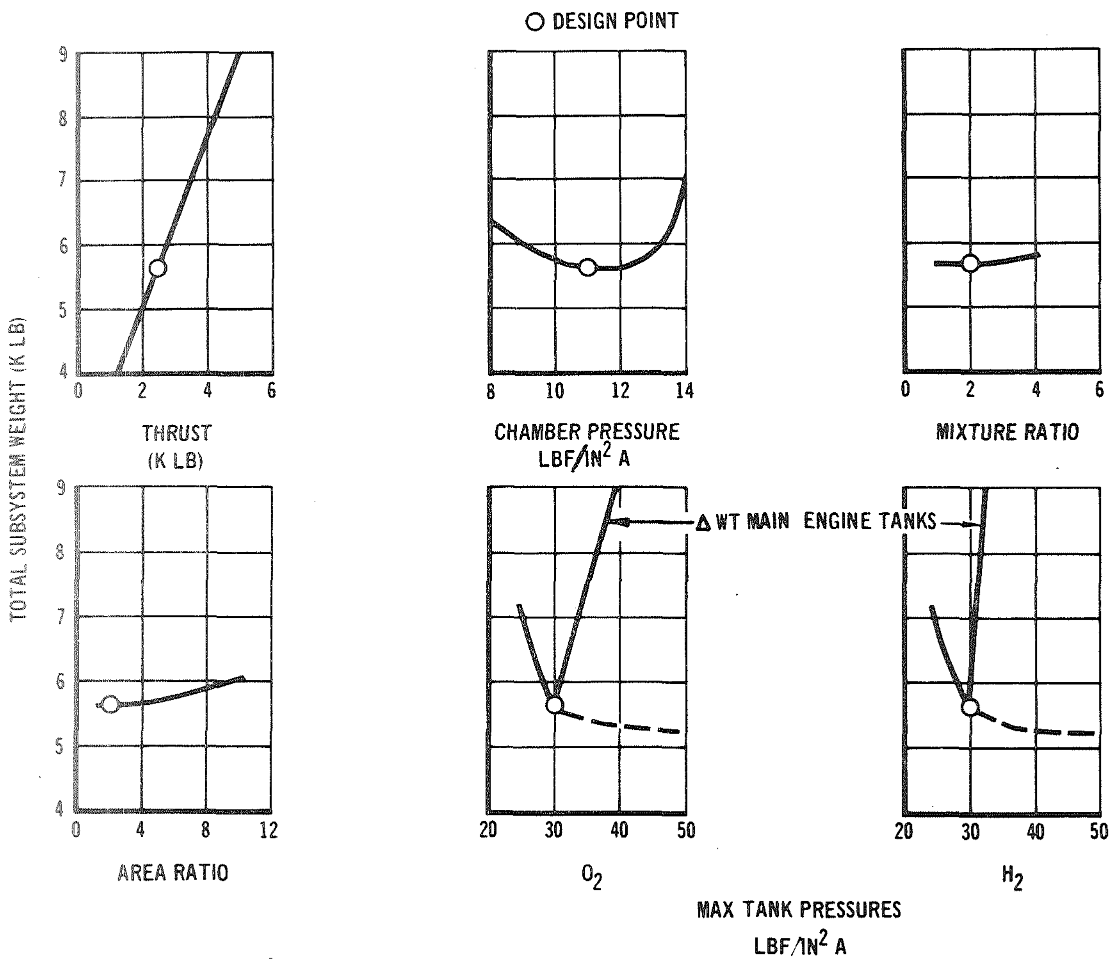
BOOSTER APS WEIGHT

FIGURE 7-1

Weight sensitivities are shown in Figure 7-2. Subsystem weight is strongly dependent on engine thrust level and chamber pressure, and is relatively insensitive to mixture ratio and expansion ratio. Subsystem weight sensitivity to main engine tank pressures is very high, if the weight penalty associated with higher pressures is included. Thus, no pressure increases above those specified in Reference (a) are recommended. The subsystem design points, based on the weight optimums shown in Figure 7-2 are:

chamber pressure	11 $lbf/in^2 a$
mixture ratio	2.0
expansion ratio	2:1

These values reflect the effect of engine performance on line sizing (in terms of flowrate requirements) and the effect of engine performance on main engine final tank pressure (as discussed earlier in section 5.2).



BOOSTER APS WEIGHT SENSITIVITIES

FIGURE 7-2

7.2 Orbiter APS - A weight breakdown of the final orbiter APS design is presented in Figure 7-3. Weights are shown for propellant, main engine propulsion modifications and each orbiter APS assembly. The propellant weight shown includes usable, vent and APS residual, but does not include the usable main engine residuals which furnish 1913 lbs of oxygen. However, the weight penalty required to utilize the main engine residuals, i.e., the compartmented tank weight penalty of 50 lb has been charged to the APS. In addition, the weight of main engine pressurant line bypass valves, required to accomodate the liquid/vapor mixer concept, have been included.

COMPONENT (NO)	WEIGHT, LB	
	O_2	H_2
PROPELLANT (*)	(4496)	(2499)
MAIN PROPULSION MODS.	(92)	(42)
COMPARTMENTED TANK	50	–
PRESSURANT LINE BYPASS VALVES (2)	42	42
PROPELLANT STORAGE ASSEMBLY	(233)	(820)
TANK, INSULATION, AND VENT	164	556
PRESSURIZATION SUBASSEMBLY	32	159
PROPELLANT ACQUISITION	37	105
PROPELLANT CONDITIONING ASSEMBLY	(149)	(327)
LINE AND MANIFOLDS	37	64
TUBING	47	98
ATTACHMENT FLANGES	56	154
VALVES (5)	9	11
LIQUID VAPOR MIXING ASSEMBLY	(96)	(108)
MIXER	17	11
THROTTLE VALVE	22	22
CONTROL VALVES (5)	44	55
REGULATORS (2)	13	20
DISTRIBUTION ASSEMBLY	(565)	(707)
LINES	174	258
COMPENSATORS, LINEAR	97	127
COMPENSATORS, ANGULAR	68	76
ISOLATION VALVES (24)	226	246
ENGINE ASSEMBLES	(2734)	
ENGINES (33)	2541	
PNEUMATIC SUBASSEMBLY		
HELIUM	11	
TANKS (3)	103	
VALVES (42)	19	
REGULATOR (5)	14	
LINES	46	
(TOTAL)	(12868)	

* INCLUDES REQUIREMENTS FOR OMS PROPELLANT SETTLING (120,000 LB-SEC).

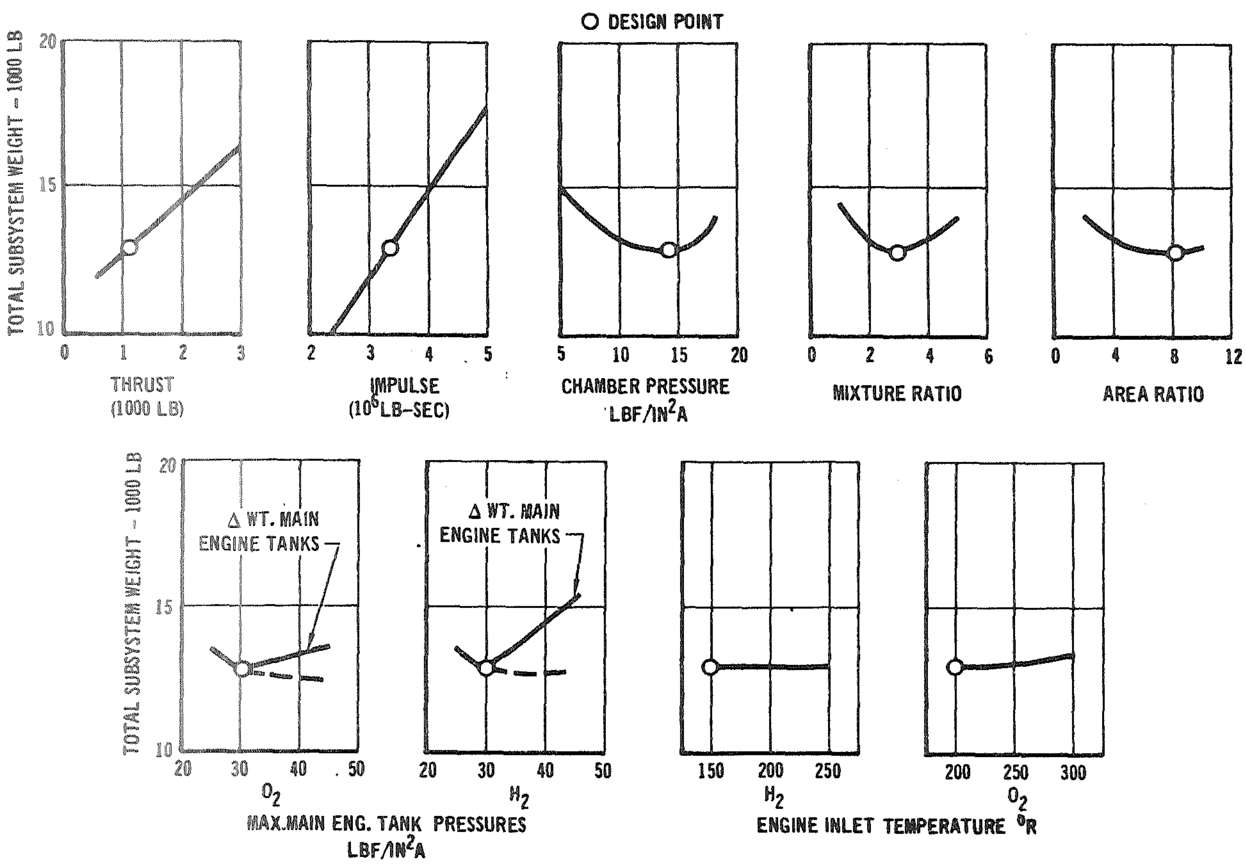
ORBITER APS WEIGHT

FIGURE 7–3

Orbiter weight sensitivities are shown in Figure 7-4. As with the booster, no weight advantage is incurred with main engine tank pressure increases, if the main engine tank weight increase associated with higher pressures is included. The subsystem is sensitive to both requirements (thrust, impulse) and subsystem design parameters (chamber pressure, mixture ratio, area ratio, and engine inlet temperature). Subsystem design parameters are:

chamber pressure	13.7 lbf/in^2a
mixture ratio	3:1
expansion ratio	8:1
engine inlet temperature	
hydrogen	150°R
oxygen	200°R

The chamber pressure, mixture ratio, and expansion ratio were selected to provide minimum subsystem weight. The engine inlet temperatures were selected as low as possible consistent with the propellant condensation temperature limits discussed in section 5.1.



ORBITER APS WEIGHT SENSITIVITIES

FIGURE 7-4

The weights shown in Figure 7-3 reflect failure criteria of nominal acceleration with <u>one</u> engine out and minimum acceleration with two engines out. A reference design point using nominal acceleration with <u>all</u> engines and minimum acceleration with two engines out has been determined to provide a common base for comparison to the high pressure APS (Reference b). The latter failure criteria allows a reduction in engine thrust level with a commensurate reduction in subsystem weight. A weight breakdown of the reference design point is shown in Figure 7-5.

COMPONENT (NO)	WEIGHT, LB O_2	WEIGHT, LB H_2
PROPELLANT (*)	(4935)	(2645)
MAIN PROPULSION MODS	(92)	(42)
COMPARTMENTED TANK	50	--
PRESSURANT LINE BYPASS VALVES (2)	42	42
PROPELLANT STORAGE ASSEMBLY	(246)	(853)
TANK, INSULATION, AND VENT	173	581
PRESSURIZATION SUBASSEMBLY	35	164
PROPELLANT ACQUISITION	38	108
PROPELLANT CONDITIONING ASSEMBLY	(149)	(327)
LINE AND MANIFOLDS	37	64
TUBING	47	98
ATTACHMENT FLANGES	56	154
VALVES (5)	9	11
LIQUID VAPOR MIXING ASSEMBLY	(96)	(108)
MIXER	17	11
THROTTLE VALVE	22	22
CONTROL VALVES (5)	44	55
REGULATORS (2)	13	20
DISTRIBUTION ASSEMBLY	(477)	(598)
LINES	150	219
COMPENSATORS, LINEAR	86	115
COMPENSATORS, ANGULAR	52	58
ISOLATION VALVES (18)	189	206
ENGINE ASSEMBLIES	(2231)	
ENGINES (24)	2052	
PNEUMATIC SUBASSEMBLY		
HELIUM	11	
TANKS (3)	103	
VALVES (42)	15	
REGULATOR (5)	14	
LINES	36	
(TOTAL)	(12799)	

* INCLUDES REQUIREMENTS FOR OMS PROPELLANT SETTLING (120,000 LB – SEC)

ORBITER APS WEIGHT

REFERENCE DESIGN POINT

FIGURE 7-5

In addition, the use of integrated APS/OMS tankage, such as that used on the high pressure APS was investigated. Tankage assembly weights have been based on the high pressure APS Subtask B report, Reference (b). Tankage and pressurization assembly weights were scaled to the APS propellant requirements and adjusted to reflect the APS pressurization requirements of 35 lbf/in^2a. The resulting subsystem weights are shown in Figure 7-6.

COMPONENT (NO)	WEIGHT, LB O_2	WEIGHT, LB H_2
PROPELLANT (*)	(4496)	(2499)
MAIN PROPULSION MODS	(92)	(42)
COMPARTMENTED TANK	50	--
PRESSURANT LINE BYPASS VALVES (2)	42	42
PROPELLANT STORAGE ASSEMBLY	(129)	(674)
TANK, INSULATION, AND VENT	82	489
PRESSURIZATION SUBASSEMBLY	32	114
PROPELLANT ACQUISITION	15	71
PROPELLANT CONDITIONING ASSEMBLY	(149)	(327)
LINE AND MANIFOLDS	37	64
TUBING	47	98
ATTACHMENT FLANGES	56	154
VALVES (5)	9	11
LIQUID/VAPOR MIXING ASSEMBLY	(96)	(108)
MIXER	17	11
THROTTLE VALVE	22	22
CONTROL VALVES (5)	44	55
REGULATORS (2)	13	20
DISTRIBUTION ASSEMBLY	(565)	(707)
LINES	174	258
COMPENSATORS, LINEAR	97	127
COMPENSATORS, ANGULAR	68	76
ISOLATION VALVES (24)	226	246
ENGINE ASSEMBLIES	(2734)	
ENGINES (33)	2541	
PNEUMATIC SUBASSEMBLY		
HELIUM	11	
TANKS (3)	103	
VALVES (42)	19	
REGULATOR (5)	14	
LINES	46	
(TOTAL)	(12618)	

* INCLUDES REQUIREMENTS FOR OMS PROPELLANT SETTLING (120,000 LB – SEC
ORBITER APS WEIGHT

INTEGRATED OMS/APS PROPELLANT TANKAGE

FIGURE 7-6

8. TECHNOLOGY CRITIQUE

Space shuttle control requirements and low pressure APS installation features dictated thrust levels of 2500 lb for the Booster and 1080 lbs for the Orbiter. Also, shuttle reuse requirements necessitate that the subsystem perform 100 missions without major refurbishment. These requirements are far beyond those of any previous control propulsion system, and no APS components capable of satisfying these requirements exist today. The purpose of this study was to develop a low pressure APS design to fulfill these requirements. The resulting low pressure APS is simple in design and operational approach. Controls for the subsystem are not complex and, at the low pressures of interest, component designs capable of satisfying both performance and life requirements appear to be relatively straightforward. Discussion of technology considerations for the APS are presented separately for each major assembly.

8.1 Propellant Storage Assembly Technology - In this assembly, there are two principal areas of technology uncertainties: (a) the pressurized, dual wall insulation assembly, and (b) design of the propellant acquisition device. Based on presently available data, high confidence can be placed in the performance of aluminized Mylar insulation with regard to its basic heat transfer characteristics. Numerous developmental programs related to high performance insulation have been conducted, and shuttle requirements do not tax the capability of this type of heat protection system. However, no programs have been conducted to demonstrate the reusability of this kind of insulation. It is known, though, that any form of condensation within Mylar layers can cause severe degradation in thermal characteristics. For this reason, the tankage system incorporates a fiberglass outer shell, pressurized during boost and entry and vented for all orbit operations. This prevents contact between insulation and atmospheric air (with resultant risk of condensation between Mylar layers). Data are available which indicate that multilayer insulation of this type will freely vent without significant pressure gradients when no protective covering is used. However, data are not available to show whether or not repeated pressurization and vent cycles would alter high performance insulation heat transfer characteristics nor to show that venting could be readily accomplished when a protective covering is used.

The principal concern in propellant storage assembly design is the propellant positioning device. Data are available which clearly demonstrate the validity of the surface tension approach for propellant positioning. However, all previous work has been conducted on relatively small scale tanks, and there has been very little experience with cryogenics. To date, the maximum size of surface tension device has been limited to approximately 1 ft in diameter. Screen production forecasts indicate a size limit in the order of 5 ft or less. Since APS propellant tanks will be 9.5 ft and 5.5 ft in diameter for hydrogen and oxygen, respectively, a novel approach to positioning will be required to accommodate screen fabrication limitations. The basic nature of surface tension screen devices prohibits the presence of gas in the contained propellant cavity, as this could result in a breakdown of liquid/gas interface (with a resultant loss of pressurant). Thus, heat leak into a cryogenic surface tension device must be minimized, since boiling within the containment device would produce vapor. An acceptable screen design consisting of ring channels to capture the propellant appears to solve the fabrication problem, and use of a vapor cooled shroud will interrupt any heat flux into the device. However, significantly more effort will be required to establish surface tension screen design data and to demonstrate achievement of successful design.

8.2 Conditioner Assembly Technology - Technology considerations associated with this assembly are related to component design and performance and to questions associated with subsystem operational analysis.

8.2.1 Conditioner Assembly Component Design - Principal concern over component design lies with the passive heat exchanger. As currently conceived, the heat exchanger consists of thin wall, aluminum tubing in intimate contact with booster tank longitudinal stiffeners. Analyses to date have not given in depth attention to the cost and difficulty of incorporating such a device on the tank. While it appears to be a simple and straightforward operation, this basic premise should be examined in detail from the viewpoint of design verification testing and tank fabrication cost.

The function of the heat exchanger is to vaporize liquid propellants from storage tanks and superheat them to required conditioning temperature. This requires a fluid phase change within the heat exchanger, and this has been classically difficult because of stability and chugging considerations. Due to the nature of the heat sink approach to conditioning, heat input will not be well controlled. When two major operations occur in a short period of time, the tank wall will be relatively cool for the second burn, resulting in a repositioning of phase change location within the heat exchanger. Thus, heat exchanger sizing will require consideration of a range of phase change positions within the heat exchanger. Test data are required to define both performance and operating characteristics of the heat exchanger. Since convective effects induced by flow velocity will certainly dominate, testing in an earth gravity environment should provide satisfactory data. Heat exchanger performance is also, to a large degree, controlled by radiation from vehicle structure to tank walls. This, of course, dictates heat exchanger recovery time. Ultimately, vehicle skin and heat protection system should be simulated in a vacuum facility, but for the present, system analysis should be relatively accurate for defining heat exchanger recovery time. Thus, simulation with the heat protection system is not seen as an immediate technology need.

8.2.2 Conditioner Assembly Analysis - Uncertainties related to analysis of the conditioner assembly center around basic thermodynamic models of the assembly during operation. There are three related areas which require more detailed investigation:

(a) motion and vaporization rates of the main engine liquid residuals after main engine cutoff; this has a large impact on the amount of APS propellant required and fixes initial boundary conditions for analyses of APS operation,

(b) mixing of resupply propellants during a major APS operation; this defines the state of propellant vapor being removed from the tanks, which, in turn, impacts heat exchangers, mixer and APS propellant requirements, and defines initial conditions for APS recovery time analyses,

(c) heat transfer characteristics within the tank following a major APS operation; these define rate of temperature and pressure recovery within the tank following a burn, and defines gas temperature profiles to be expected at start of the next burn.

Liquid Residuals Effects - APS analyses to date have considered only nominal, trapped liquid residuals. This is the minimum quantity, hence the most conservative assumption from the standpoint of subsystem weight definition. At main engine shutdown, approximately 3200 lb of liquids will be trapped. Based on extrapolation of Saturn S-IVB data, vaporization rates have been forecast. These, coupled with APS mission profiles, indicate that only 610 lb (or 19 percent) of this could actually be used by the APS (without modification of main engine tanks).

Two additional aspects not fully covered by APS studies should be considered: (1) control of vaporization to make more liquid available to the APS (resulting in a very large potential weight savings), and (2) the fact that since orbiter engines will be shutdown on a velocity command, rather than upon propellant depletion, much more liquid could be onboard at shutdown.

Regarding control of vaporization, extrapolated S-IVB data identify an equivalent liquid position or wetted wall area. Based on this area, vaporization rates are calculated for particular vehicle/tank interface and mission characteristics. Liquid position is of primary importance to the analyses but because of the low gravity environment, and multiaxis control accelerations, it is very difficult to define. The current method of using extrapolated S-IVB data is not an adequate base for subsystem design.

Secondly, since wetted wall area is of predominant importance, knowledge of the amount of residual liquid is a key factor. As previously stated, this information will not be known with accuracy for the orbiter.

Two approaches to resolution of these factors are possible: (1) control of liquid quantity and/or contact area, and (2) knowledge of liquid position and effects of control accelerations on position. The former approach was elected for the orbiter and the latter for the booster. Additional effort is required to define residual containment control requirements (orbiter) and fluid dynamic effects (booster).

Vapor Mixing Analysis - During each major APS burn, propellant is extracted from the main engine tank and simultaneously resupplied at the same rate. The temperature of propellant extracted defines tank pressure decay rates and the percentage of liquid supplied in the mixing assembly. When the temperature of gas being removed from the booster tank is high, a large quantity of liquid can be used for a prescribed impulse demand. Thus, since the propellant is resupplied at a rate equal to extraction rate, temperature of the gas being extracted also defines passive heat exchanger flow rate and amount of tank wall cooling which will occur during burn. Current analyses assume that gas is extracted from a tank whose contents are always at a homogeneous temperature. Additional studies, described in Appendix B, indicate that thermal stratification and withdrawal of cool vapors is desirable. Thus, the resupply vapors should be added to the main engine tank without significant mixing. However, there are neither sufficient data nor valid analytical techniques to establish with confidence the injection-mixing-extraction characteristics within the main engine tank. To develop an understanding of processes within the tank, a means of flow-mixing visualization is required. Scaling laws which would allow simulation of these affects in an earth gravity environment should be developed to provide this comprehension.

Heat Transfer Analysis - Basic APS flexibility is controlled by main engine tank heat transfer characteristics. After a major APS operation, tank walls are chilled and the tank is filled with relatively cool vapor. Radiative and conductive heat transfer from the vehicle structure govern assembly recovery time. Part of the heat transferred to the tank goes into raising tank wall temperature, while the remainder is given up to the vapor. The model used in this study is described in Appendix B. Two features of this model are important to APS performance: (1) it defines design of the heat exchanger, and (2) it defines thermal stratification characteristics prior to the next major APS operation.

Regarding the first, the total heat absorbed by the tank/gas system is not strongly effected by internal tank to gas heat transfer. However, if rates are not known, the balance of heat addition will not be known and the design will have to be overly conservative to accommodate uncertainties. Stated another way, the heat exchanger will have to be over-sized for the lowest tank temperature, while propellant supply and distribution will have to be over-sized for lowest gas temperature and pressure in the tank. In the extreme case of near zero heat transfer (gas highly stratified) this could result in a 400 lb increase in APS weight.

In regard to thermal/stratification, it is necessary to consider the condition of the gas that will be extracted during the next major APS operation. Under conditions of low acceleration, thermal stratification will occur in the tank. In general, this can be expected to result in a cool propellant core in the tank center and warmer propellant next to the tank walls. Three dimensional effects, and tank temperature variations, will, of course, effect these gradients (as will control moments). The degree of stratification will also have a strong impact on heat transfer characteristics. A means of simulating this phenomenon in an earth environment is needed to guide heat transfer analysis and to establish initial conditions for large propellant extractions.

8.3 Liquid/Vapor Mixer Technology - As currently conceived, the liquid/vapor mixer assembly consists of two independent controls: (1) pressure regulator, and (2) liquid flow control. In addition, there is the injection/mixing chamber itself. All of these are relatively simple in concept, but require technology extensions for low pressure APS application.

The pressure regulator is fundamentally an iris (or camera shutter) design with a motor drive to control aperture diameter. Devices such as this have been used in other applications where response requirements were much lower. Designs which will provide a long life and fast response must be developed.

The liquid flow control is a throttle valve, also motor driven, and its response must be of the same order as that of the pressure regulator. Again, while no component is currently available, similar controls have been used and no significant or long-lead development activity is considered essential.

Controls for both pressure regulator and throttle valve are similar. Each will be driven to provide a preselected measurement and, for stability, valve position rate damping will be provided by definition of the number of engines firing. For the regulator, pressure transducers of sufficient accuracy can be readily developed. For the liquid control, measurement of temperature is required. Alternate transducing methods (such as resistance thermometers or thermocouple piles) should be investigated further, to define achievable accuracy and reliability.

Preliminary analyses of the injector/mixer have shown that the O_2 vapor/liquid mixer is the more critical. This is true primarily because oxygen saturation temperature is much closer to vapor temperature, and hence has a much lower driving temperature for vaporization than hydrogen. Also, these preliminary analyses have shown that very fine spray injection techniques will be required to achieve reasonable mixing/vaporization lengths. The analyses have been performed using techniques which, while valid at high gas/liquid temperature differences, have not been verified at the low temperature differences of concern. Therefore, early testing of the oxygen mixer should be undertaken to ascertain injector performance and mixing lengths, and to verify analytical techniques.

Subsequent to development of basic injector/mixer parametric performance data, a breadboard assembly coupling regulator, throttle valve and controls should be assembled and tested to confirm stability, establish response and accuracy, and define assembly interface characteristics. This mockup should give particular attention to liquid supply line and assembly mounting to determine effect of vapor formation in the liquid line ahead of the injector.

8.4 Engine Technology - Engine assemblies for low pressure APS were one of the most obvious concerns when the concept was initially identified. For this reason, exploratory development programs on both engines and their propellant valves were initiated by the NASA. To date these programs, and the work of several propulsion companies, have shown that performance goals, cooling, and ignition can be accomplished. Further test and design effort on the engines should stress certain subsystem aspects that have been identified by APS conceptual definition studies. Specifically, these include:

(a) Minimum weight - due the low pressure under consideration and relatively high thrust levels, engines are large and constitute a significant fraction of APS inert weight. Designs should stress light weight materials and fabrication techniques for bimetallic construction.

(b) Low temperature propellants - Temperature of propellants acceptable for engine operation is a key APS design constraint. Significant APS weight savings can be realized by reducing propellant inlet temperature requirements. Based on current subsystem analyses, the limiting factor in this is hydrogen and oxygen temperatures required to prevent O_2 condensation ahead of injectors. Preliminary analyses of this thermal interchange have been performed, but have not been demonstrated. Low pressure engine designs should, in the future, stress ability to operate steady-state with propellant inlet temperatures as low as possible.

(c) Sensitivity of inlet conditions - Current APS design will closely control propellant inlet conditions during any significant APS activity. Under these conditions, vapor temperatures will be at their lowest values and pressure will be constant. However, during low usage control operations, both temperature and pressure at the engine inlet will vary over wide ranges. Since burn durations will be short, and performance is not critical during these periods, no particular problem is anticipated. However, engine design and cooling margins must be sufficiently flexible to accommodate these inlet variations.

9. SUBSYSTEM RELIABILITY

The APS has been designed to satisfy fail operational/fail safe requirements as specified in the SSVDRD. The baseline shuttle low pressure APS, as described in this report, retains a basic simplicity which will enhance the attainment of a high level of operational reliability and safety. A reliability evaluation was conducted to ensure that the subsystem contains sufficient redundancy to meet this requirement, and to provide a numerical estimate of operational subsystem reliability.

Three basic reliability tools employed in evaluating the low pressure APS baseline design have included failure mode and effect analysis, functional flow diagrams, and reliability estimates. These analyses are presented in Appendix C. Component failure rates used in deriving reliability estimates are listed in Figure 9-1 together with component duty cycles and failure modes considered.

Results of the reliability estimate are listed in Figure 9-2 for each functional group of components and for both orbiter and booster subsystems. Limiting assumptions are included in the table. Results indicate that the orbiter APS baseline operational reliability for a 72 hour earth-orbit mission will be 0.994 with fail-safe reliability exceeding 0.9999. The booster APS baseline shows an operational reliability of approximately 0.999 and a fail-safe reliability greater than 0.9999999.

Reliability estimates for both orbiter and booster APS subsystems are sensitive to change in engine failure rate. This occurs because of the relatively high number of firing cycles required of each engine, and from the fact that full triple redundancy of engines is not provided. There is a degree of conservatism in the estimate for the engines, especially in the fail-safe calculations because it was assumed that a failed open engine would be isolated and not used again. In reality, an engine that is leaking propellant could be used to provide necessary thrust in an emergency.

SHUTTLE LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM
COMPONENT DUTY CYCLES FAILURE MODES AND FAILURE RATES
ORBITER AND BOOSTER

COMPONENT TYPE	IDENTIFICATION NUMBERS (SCHEMATIC)	MISSION DUTY CYCLE	FAILURE MODE	FAILURE RATE X 10^6	SOURCE OF BASIC FAILURE RATE
LIQUID-VAPOR MIXER		33 CYCLES 800 SECONDS	INJECTOR CLOGS	NEGLIGIBLE	---
PUMP, CRYOGENIC, MOTOR DRIVEN	CRYOPUMP #1 CRYOPUMP #2 CRYOPUMP #3	123 CYCLES 900 SECONDS	INOPERATIVE	5.0/CYCLE + 50/HOUR	CYCLIC - ESTIMATE HOURLY - PESCO PRODUCTS DATA
REGULATOR, PRESSURE, HELIUM	R-1, R-3, R-5, PR-1, PR-2, PR-3, PR-4, PR-5	72 HOURS (ORBITER) 0.1 HOURS (BOOSTER)	FAILS OPEN	30/HOUR	MDAC GEMINI EXPERIENCE
			FAILS CLOSED	11/HOUR	
REGULATOR, PRESSURE, MOTOR DRIVEN IRIS VALVE	R-7, R-9	33 CYCLES 800 SECONDS	FAILS IN LOW FLOW POSITION	50/HOUR 25/CYCLE	BASIC FAILURE RATE ASSUMED THE SAME AS FOR MOTOR DRIVEN THROTTLING VALVE
			FAILS IN HIGH FLOW POSITION	50/HOUR 25/CYCLE	
VALVE, CHECK LH_2	CV-2, CV-4, CV-6, CV-8, CV-10, CV-12	123 CYCLES EACH	FAILS OPEN	2.2/CYCLE	AVCO RELIABILITY ENGINEERING DATA SERIES, FAILURE RATES, APRIL 1962
			FAILS CLOSED	0.1/CYCLE	
VALVE, S/O, PNEUMATIC ACTUATION, NORMALLY CLOSED, FILL & DRAIN	V-1, V-2, V-3, V-4, V-10, V-12, V-14, V-19, V-21	1 CYCLE EACH	FAILS OPEN	510/CYCLE	FARADA - SATURN
			FAILS CLOSED	100/CYCLE	
VALVE, S/O, MOTOR DRIVEN, ISOLATION	V-5, V-15, V-43, V-45, V-47, V-49, PLUS ALL ENGINE ISOLATION VALVES	1 CYCLE EACH AS REQUIRED	FAILS OPEN	16/CYCLE	NASA DATA-SATURN (SHUTOFF VALVE)
			FAILS CLOSED	3.3/CYCLE	
VALVE, LIQUID THROTTLING MOTOR DRIVEN	TV-1, TV-3	33 CYCLES 800 SEC	FAILS OPEN	80/HOUR + 40 CYCLE	HOURLY - LTV ELECTROSYSTEMS DATA
			FAILS CLOSED	20/HOUR + 10 CYCLE	CYCLIC - STANFORD RESEARCH INSTITUTE CONTRACT NAS 7-751 (ELECTROMECHANICAL ACTUATOR)
VALVE, SOLENOID ACTUATED, HELIUM CONTROL	V-6, V-7, V-9, V-11, V-13, V-15, V-25, PV-1, PV-2, PV-3, PV-4, PV-5, PV-6, PV-7, PV-8, PV-9	1 CYCLE EACH AS REQUIRED	FAILS OPEN	8.0/CYCLE	THIOKOL CHEMICAL CORP. (SURVEYOR)
			FAILS CLOSED	0.77/CYCLE	
	ENGINE PILOT VALVES AND BACK-UP VALVES	SAME AS THRUSTERS			
VALVE, SOLENOID ACTUATED, LIQUID SHUTOFF	V-31, V-37, V-39, V-41	1 CYCLE EACH AS REQUIRED	FAILS OPEN	5.8/CYCLE	NASA DATA - SATURN
	V-29, V-33	123 CYCLES (H_2) 87 CYCLES (O_2)	FAILS CLOSED	0.56/CYCLE	
	V-35, V-27	18 CYCLES - REENTRY			
VALVE, RELIEF, PRESSURE	RV-1, RV-2, RV-3, PRV-1, PRV-2, PRV-3, PRV-4	1 CYCLE EACH AS REQUIRED	FAILS OPEN	5.3/CYCLE	NASA DATA - SATURN
			FAILS CLOSED	1.5/CYCLE	
(BURST DISK)	RV-1, RV-2, RV-3, PRV-1, PRV-2, PRV-3, PRV-4		FAILS OPEN	60/UNIT	STANFORD RESEARCH INSTITUTE CONTRACT NAS 7-751
			FAILS CLOSED	40/UNIT	
TANK ASSEMBLY, LIQUID PROPELLANT	LO_2 TANK LH_2 TANK	72 HOURS	FAILS TO MAINTAIN TEMPERATURE CONTROL	NEGLIGIBLE	---
ENGINE ASSEMBLY FILM COOLED (INCLUDING VALVES)	ORBITER + X - X PITCH & ROLL YAW	24 CYCLES 9 CYCLES 650 CYCLES 650 CYCLES EACH ENGINE	FAILS TO FUNCTION	5.0/CYCLE	MDAC ESTIMATE-AFTER DISCUSSION WITH ENGINE MANUFACTURERS (AEROJET, BELL, MARQUARDT, AND ROCKETDYNE)
	BOOSTER PITCH & ROLL YAW	100 CYCLES 40 CYCLES EACH ENGINE	FAILS OPEN OR LEAKS	7.5/CYCLE	
QUANTITY MEASURING DEVICE	PQ-1, PQ-2, PQ-3, PQ-4	72 HOURS (ORBITER) 0.1 HOURS (BOOSTER)	FAILS TO FUNCTION	2.5/HOUR	AVCO RELIABILITY ENGINEERING DATA SERIES, FAILURE RATES, APRIL 1962

FIGURE 9-1

FUNCTIONAL GROUP	ORBITER OPERATIONAL	ORBITER FAIL SAFE	BOOSTER OPERATIONAL	BOOSTER FAIL SAFE
LO_2 STORAGE AND PRESSURIZATION	0.999990	0.99999997	-	-
LH_2 STORAGE AND PRESSURIZATION	0.999997	0.99999999	-	-
PROPELLANT CONDITIONING - O_2	0.999994	0.99999999	-	-
PROPELLANT CONDITIONING - H_2	0.999993	0.99999999	-	-
PROPELLANT DISTRIBUTION AND ENGINES	0.999743	0.99997069	0.999923	0.99999998
ENGINE PNEUMATIC CONTROL	0.999749	0.99999996	0.999991	0.99999999
SUBSYSTEM	0.994118	0.99991134	0.999914	0.99999997

ASSUMPTIONS:
(1) COMPONENT EXTERNAL LEAKAGE CAN BE CONTROLLED BY PROPER SEAL DESIGN
(2) SENSING AND SWITCHING RELIABILITY IS EQUAL TO 1.0.
(3) STRUCTURAL RELIABILITY IS EQUAL TO 1.0.
(4) MAIN PROPULSION SUBSYSTEM COMPONENTS USED BY THE AUXILIARY PROPULSION SYSTEM WILL NOT DEGRADE APS OPERATION OR SAFETY.
(5) THE NON-OPERATING FAILURE RATE FOR APS COMPONENTS WILL NOT BE SIGNIFICANT

APS RELIABILITY

FIGURE 9-2

10. SUMMARY AND CONCLUSIONS

Subtask B consisted of preliminary design of the APS concept selected during Subtask A. The study concentrated on two primary areas, components and subassembly designs, and subsystem performance and optimization. The first resulted in definition of: the propellant storage subassembly, heat exchanger mounting, main engine tank propellant acquisition, feed line sizing and routing, isolation valve design and engine thrust level, number of engines and installation restrictions. Feed line, isolation valve, and engine subassemblies were optimized to provide minimum weight penalty. This optimization involved alternate engine locations and number of engines, isolation valve concept selection, definition of line routing and the number of isolation valves required, and sizing optimization to define engine chamber pressure, and line and valve pressure losses. The second area of concentration, involving performance definition and optimization, resulted in: definition of heat exchanger and liquid/vapor mixer operating characteristics, definition of main engine tank temperature and pressure histories, and identification of subsystem performance and propellant requirements, including the effect of main engine propellant residuals.

(a) Orbiter - Propellant storage consisted of insulation, acquisition, storage and pressurization subassemblies. The design selected used aluminized mylar high performance insulation (protected by an outer fiberglass jacket) optimized for minimum storage weights; a passive, surface tension acquisition concept consisting of several screen covered channels arranged in layers; and an active hydrogen vent/cooling subsystem to intercept the incoming heat flux. Cold helium oxygen pressurization was used, but hydrogen pressurization was best satisfied by using a tank sump mounted boost pump with helium prepressurization to 40 lbf/in^2a.

The tank mounted passive heat exchanger was attached to the tank ribs by riveting. The increased rib section modulus of this concept allowed a reduction in oxygen tank rib height, thus reducing heat exchanger weight penalty. Heat exchanger design characteristics, as well as those of the liquid/vapor mixer, were defined during performance evaluation.

Orbiter APS mission requirements were met with an engine arrangement utilizing thirty-three 1080 lb thrust engines operating at a chamber pressure of 13.7 lbf/in^2a.

A dual mode of operation, using liquid/vapor mixing during major maneuvers and simple blowdown during low usage demonstrably provides improved performance and reduced technology requirements. Optimum APS/OMS velocity allocation was defined, with the OMS providing the few (three to four) largest burns, while the APS provides the remaining functions. This allocation provides reduced weight penalty without excessive OMS requirements in terms of number of starts and burn duration. With this allocation, the APS was shown to provide high performance throughout the missions.

The APS design as defined during this study provides high performance and high reliability and satisfies all of the design objectives. Increased technology requirements were identified only with main engine tank thermodynamics and propellant storage and acquisition. No significant problems are anticipated, but additional analytical and test requirements have been identified.

(b) Booster - The booster APS requires no separate propellant storage, since propellant residuals, trapped in the main engine tanks following boost, are sufficient to meet APS propellant demands. The booster APS operates in a simple blowdown mode and no additional control is required. Booster APS mission requirements were met with an engine arrangement utilizing twenty 2500 lb thrust engines operating at a chamber pressure of 11 lbf/in^2a.

11. REFERENCES

a) Space Shuttle Vehicle Description and Requirements Document (NASA), dated 1 October 1970.

b) Gaines, R.D., Goldford, A.I., Kaemming, T.A., High Pressure Auxiliary Propulsion Subsystem Definition, Subtask B Report, McDonnell Douglas Report No. MDC E0298, dated 12 February 1971.

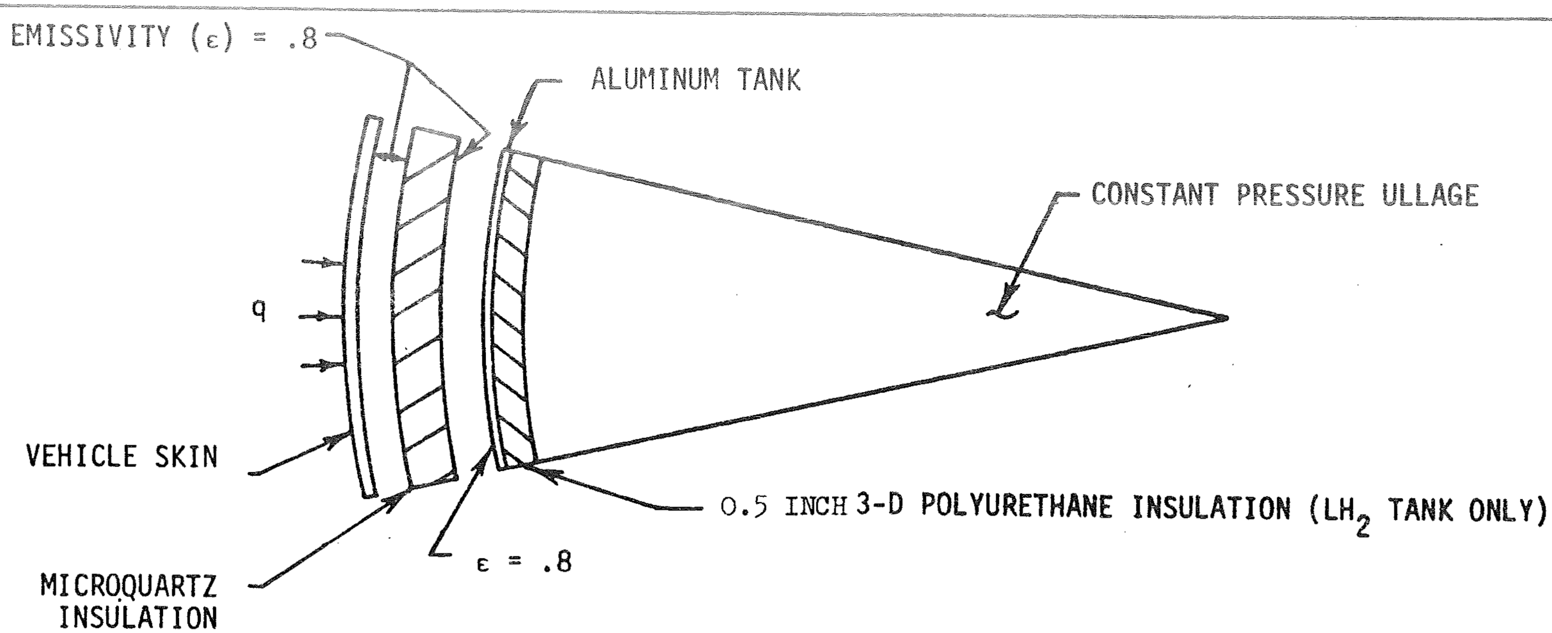
APPENDIX A

THERMAL ENVIRONMENT DEFINITION

A-1. INTRODUCTION

APS performance depends strongly on thermal environment. Heat transfer to main engine tanks determines rate of residual liquid boiloff, performance and response characteristics of tank mounted heat exchangers, as well as main engine tank temperature and pressure histories during the mission. The following paragraphs describe the model used to evaluate main engine tank thermal environment. The effect of this environment on APS performance in terms of propellant requirements and tank pressure fluctuations is described in Appendix B

The model used to calculate heat transfer to main engine tank is shown in Figure A-1. It consists of an external skin; a radiation gap; a layer of Micro-Quartz; another radiation gap; the main engine tank wall; and, for the LH_2 tank, an internal layer of polyurethane foam insulation. To establish the thermal environments for APS operating analysis transient ascent and on-orbit temperatures were calculated, using appropriate ascent and orbital heating rates. Values shown for skin, Micro-Quartz, and tank properties corresponded to those obtained from MDAC Phase B study for mid-tank locations.



a) MODEL SCHEMATIC

PROPELLANT TANK	LOCATION ON TANK	VEHICLE SKIN THICKNESS	VEHICLE SKIN MATERIAL	MICRO-QUARTZ INSULATION THICKNESS	TANK THICKNESS
HYDROGEN	BOTTOM	.047	Cb - 752	2.0 INCHES	.058 INCH
HYDROGEN	SIDE	.023	RENE'41	0.2 INCHES	.058 INCH
OXYGEN	BOTTOM	.047	Cb - 752	2.0 INCHES	.084 INCH
OXYGEN	SIDE	.023	RENE'41	0.2 INCH	.084 INCH

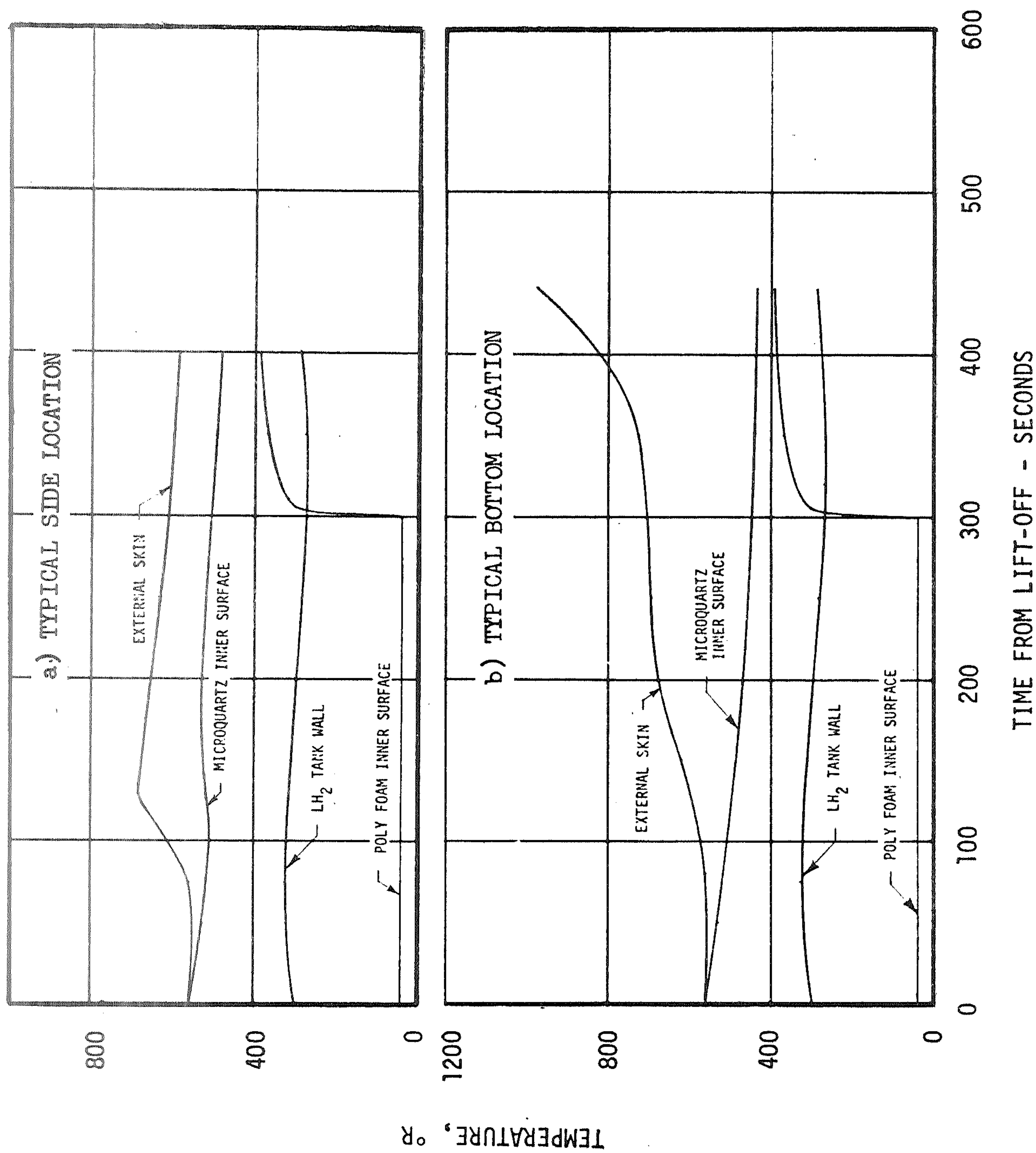
b) MODEL COMPONENT THICKNESSES

THERMAL MODEL

FIGURE A-1

A-2. ORBITER ASCENT HEATING

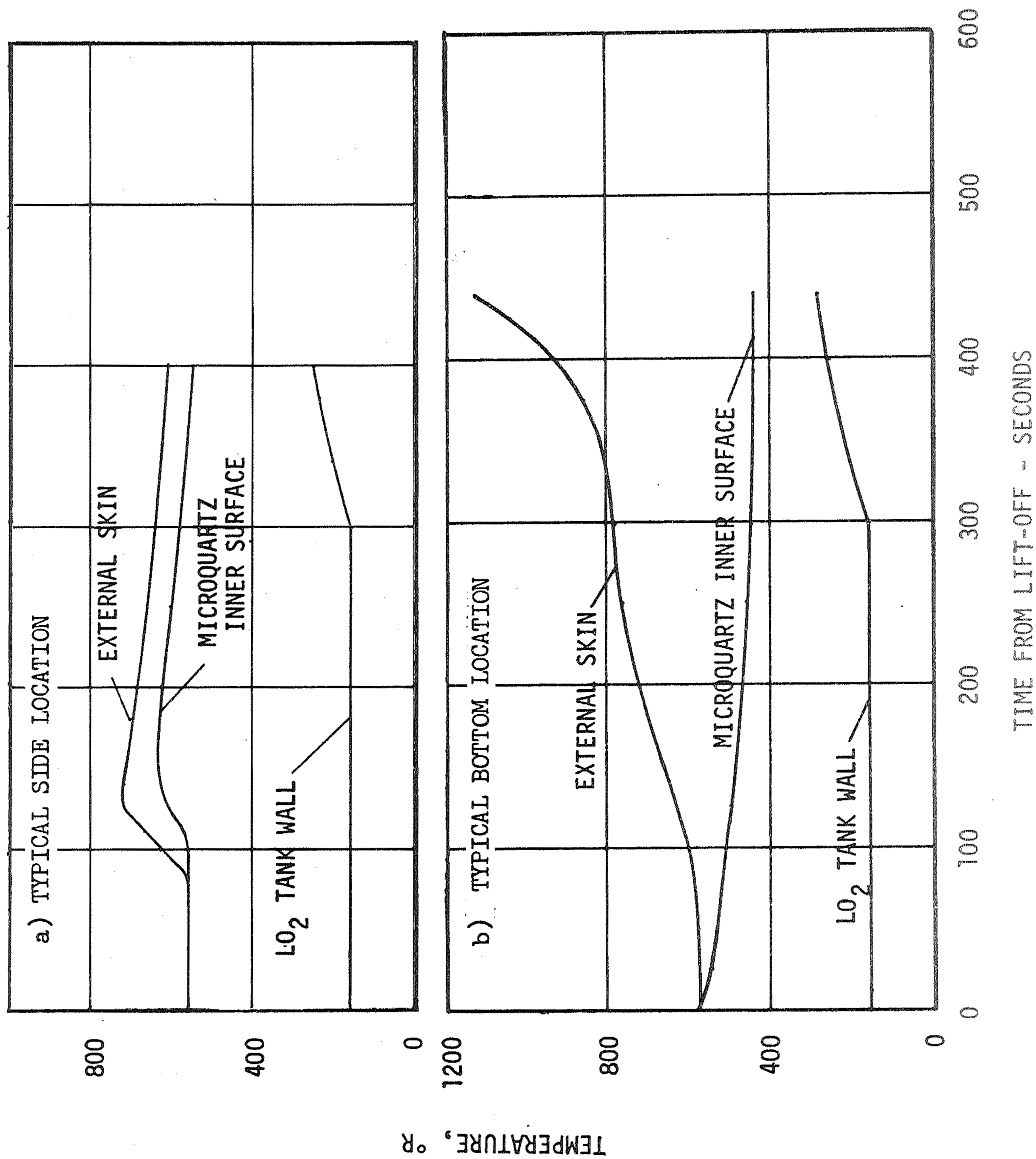
Typical LH_2 tank side and bottom temperature histories during ascent are shown in Figure A-2, based on an initial tank wall temperature of 300°R. LO_2 tank temperature histories are shown in Figure A-3. Comparison of LO_2 tank wall history with LH_2 tank wall history reveals the moderating effect of internal polyurethane insulation installed within the LH_2 tank. The LO_2 tank incurs substantial heating from heated pressurant gases, and begins to increase in temperature, whereas LH_2 tank temperature is almost constant after propellant uncovers the internal tank wall.



LH_2 TANK ASCENT TEMPERATURE PROFILES

PRESSURANT TEMP= 430°R

FIGURE A-2



LO_2 TANK ASCENT TEMPERATURE PROFILES

FIGURE A-3

A-3. ORBITER ON-ORBIT HEATING

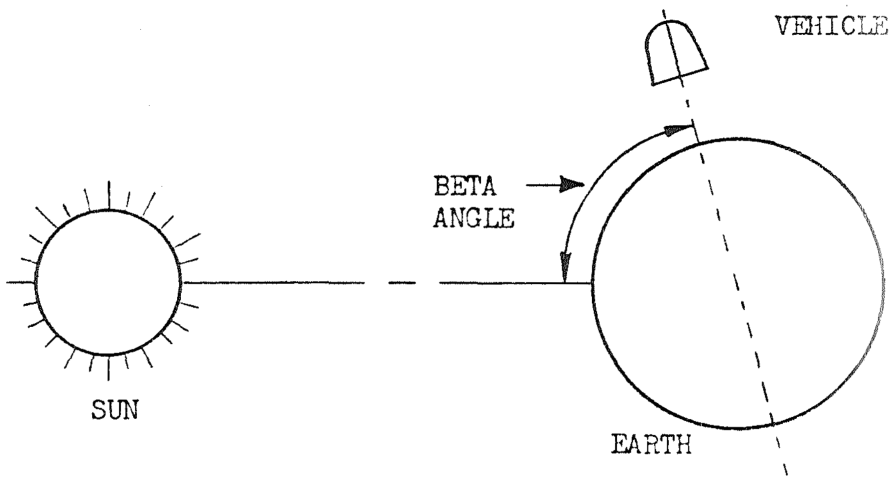
During orbit, difference in heat flux associated with different vehicle locations and orbital trajectories leads to substantial variations in vehicle internal environment. As a simplification, the orbiter bottom is assumed always to face the earth for these analyses and a range of sun-earth-vehicle angles as illustrated in Figure A-4 was considered. Envelopes of orbiter component environmental temperatures are presented in Figure A-5 for high and low beta angle orbits. For the low beta angle case, three regions have been defined, corresponding to the temperature history expected (1) for the inner surface of the Micro-Quartz on the top of the vehicle, where oscillations are most severe, (2) at the side, where the heat flux is comparatively low, and (3) on the bottom, where thick Micro-Quartz and relatively constant heat flux from the earth maintain a nearly constant inner surface temperature. Depending upon their location, components are exposed to either an almost constant temperature (corresponding to side or bottom location) or temperature oscillations (similar to those of the top).

For the high beta angle case, where a hot and cold vehicle side may be identified, the environmental range shown corresponds to that expected for the first orbit as well as for steady-state conditions. The first orbit upper limit is the temperature of the inner surface of the Micro-Quartz insulation on the hot side.

Calculations of temperature histories for the elements of the model shown in Figure A-1 for the cold side, using only the cold side external heating rates, predict temperatures which are unrealistically low. A better estimate of the cold side environment has been obtained by assuming that the cold side is heated from the hot side via a radiation shield, e.g., the internal keel web. The first orbit lower limit environment temperature shown is the temperature of such an intermediate shield, except for a short period just after insertion. During this time, the vehicle is cooling from ascent heating and the inner surface of the cold side Micro-Quartz is used for the environmental temperature. The composite cold side curve should thus represent the nominal temperature level any component on that side will experience.

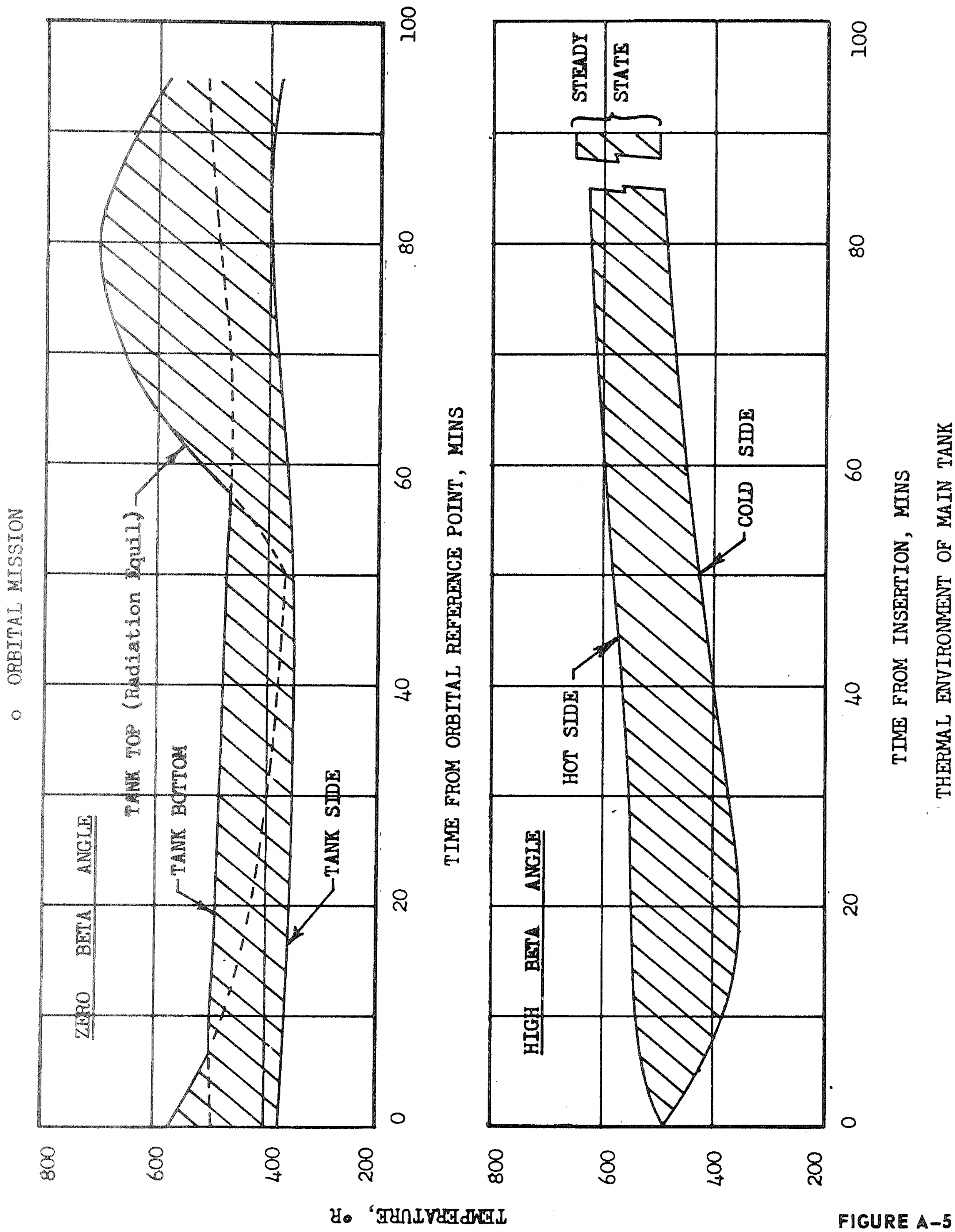
The limits of the steady-state range are the Micro-Quartz inner surface temperatures for a steady-state one dimensional calculation, using appropriate external heat fluxes and permitting radiation from hot to cold side.

Heat transfer characteristics associated with tank mounted heat exchangers which are buried within the vehicle skin were evaluated. Figures A-6 and A-7 show typical results. Figure A-6 illustrates the tank structural temperature along the tube length at various times during a burn and shows the variance in temperature experienced at the end of a burn. Using these initial conditions, recovery temperatures were evaluated for representative heat fluxes. These results are shown in Figure A-7. As shown the temperature recovery of tank mounted heat exchangers does not appear to be a significant problem.



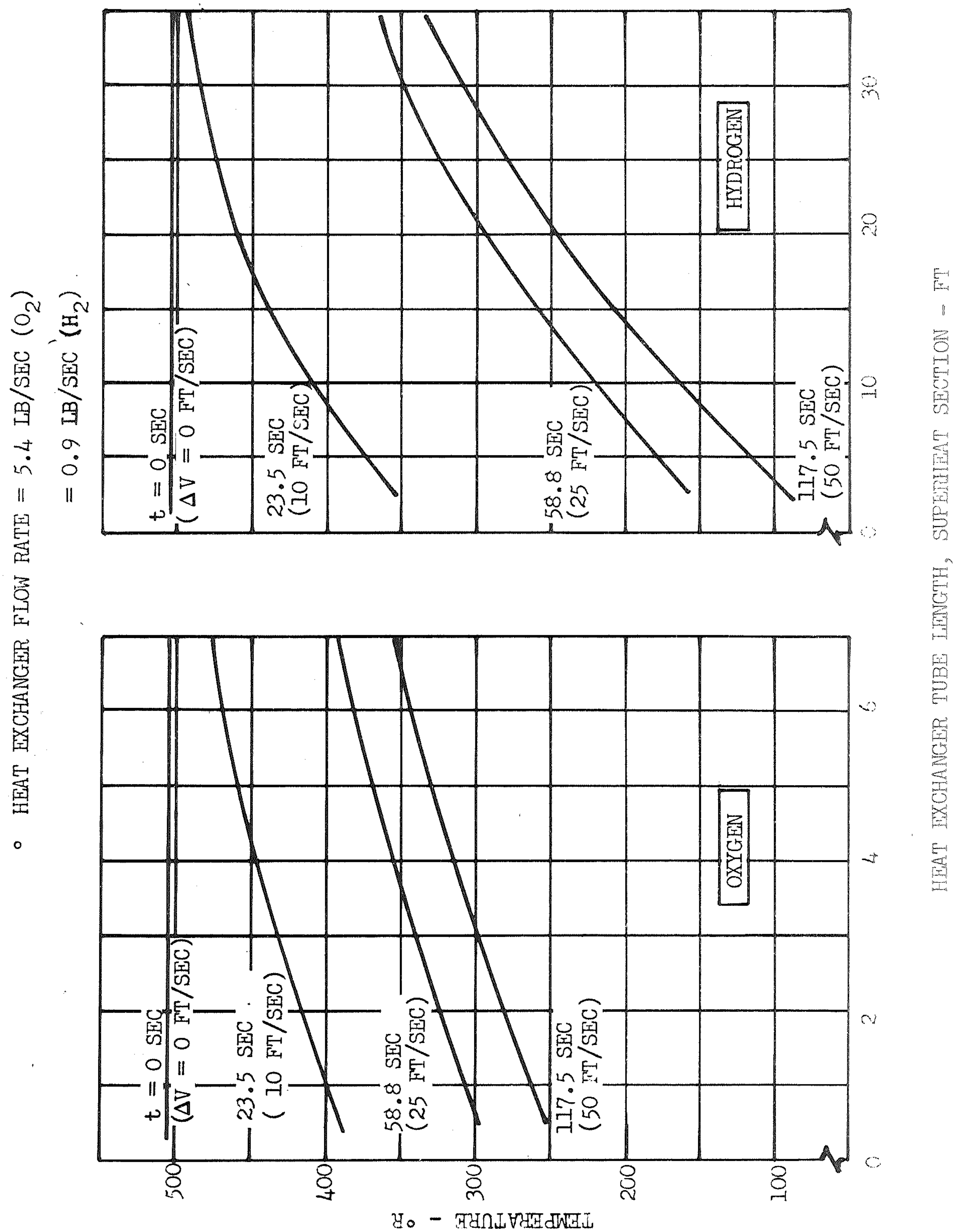
SCHEMATIC OF SUN-EARTH-VEHICLE (BETA) ANGLE

FIGURE A-4



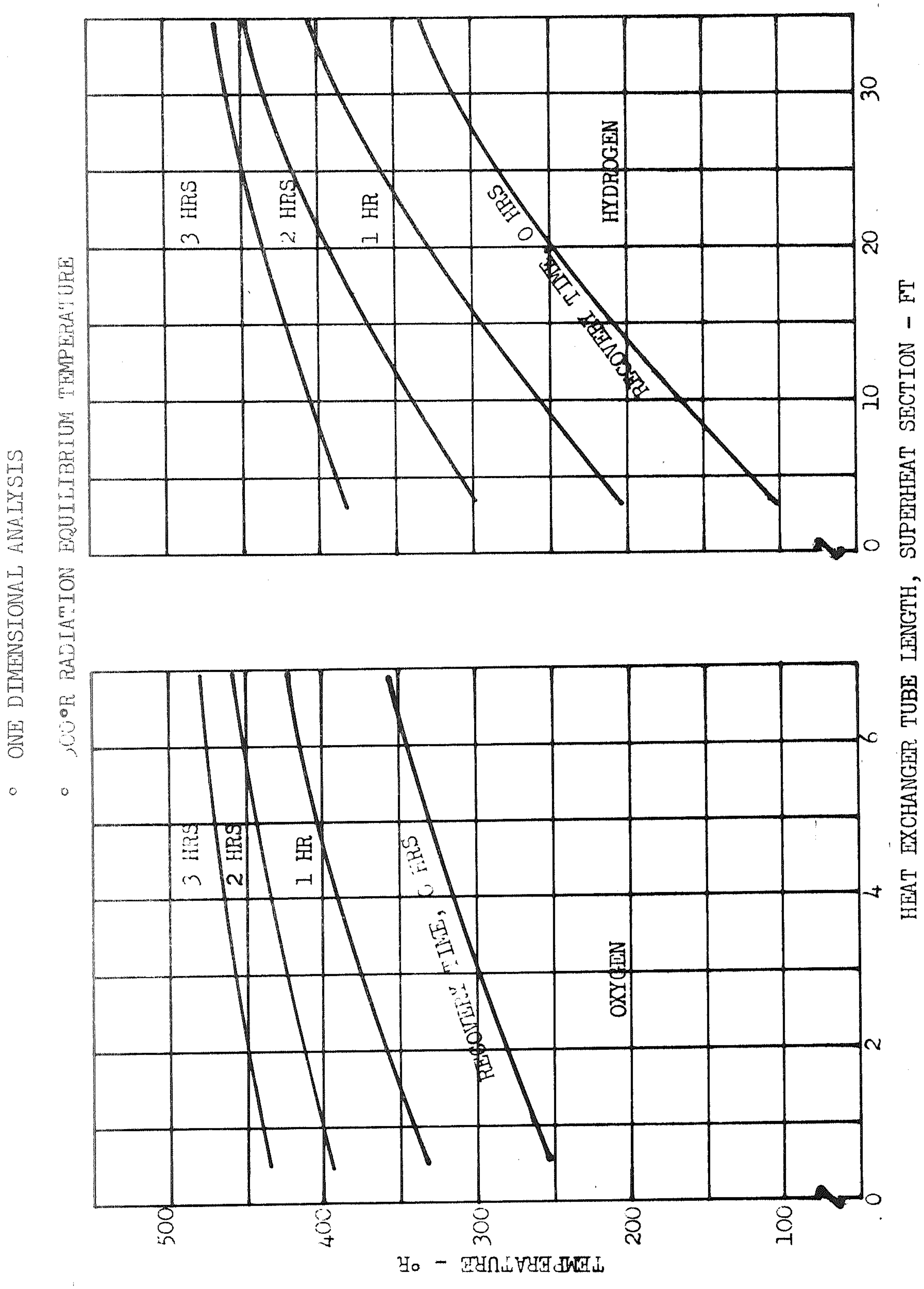
THERMAL ENVIRONMENT OF MAIN TANK

FIGURE A-5



HEAT EXCHANGER TEMPERATURE
DURING +X MANEUVER

FIGURE A-6



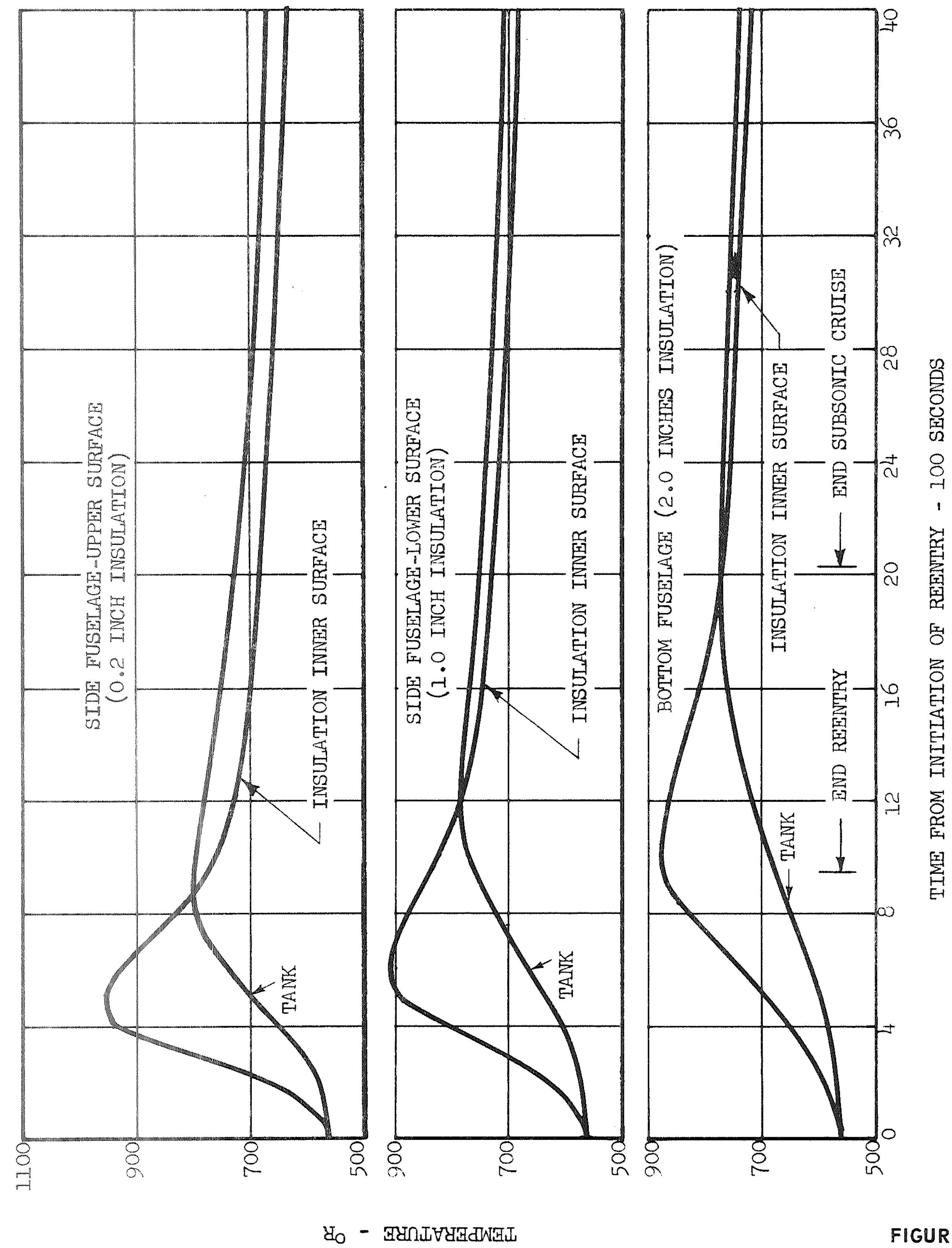
HEAT EXCHANGER TEMPERATURE
RECOVERY FOLLOWING 50 FPS +X MANEUVER

FIGURE A-7

A-4. ORBITER REENTRY HEATING

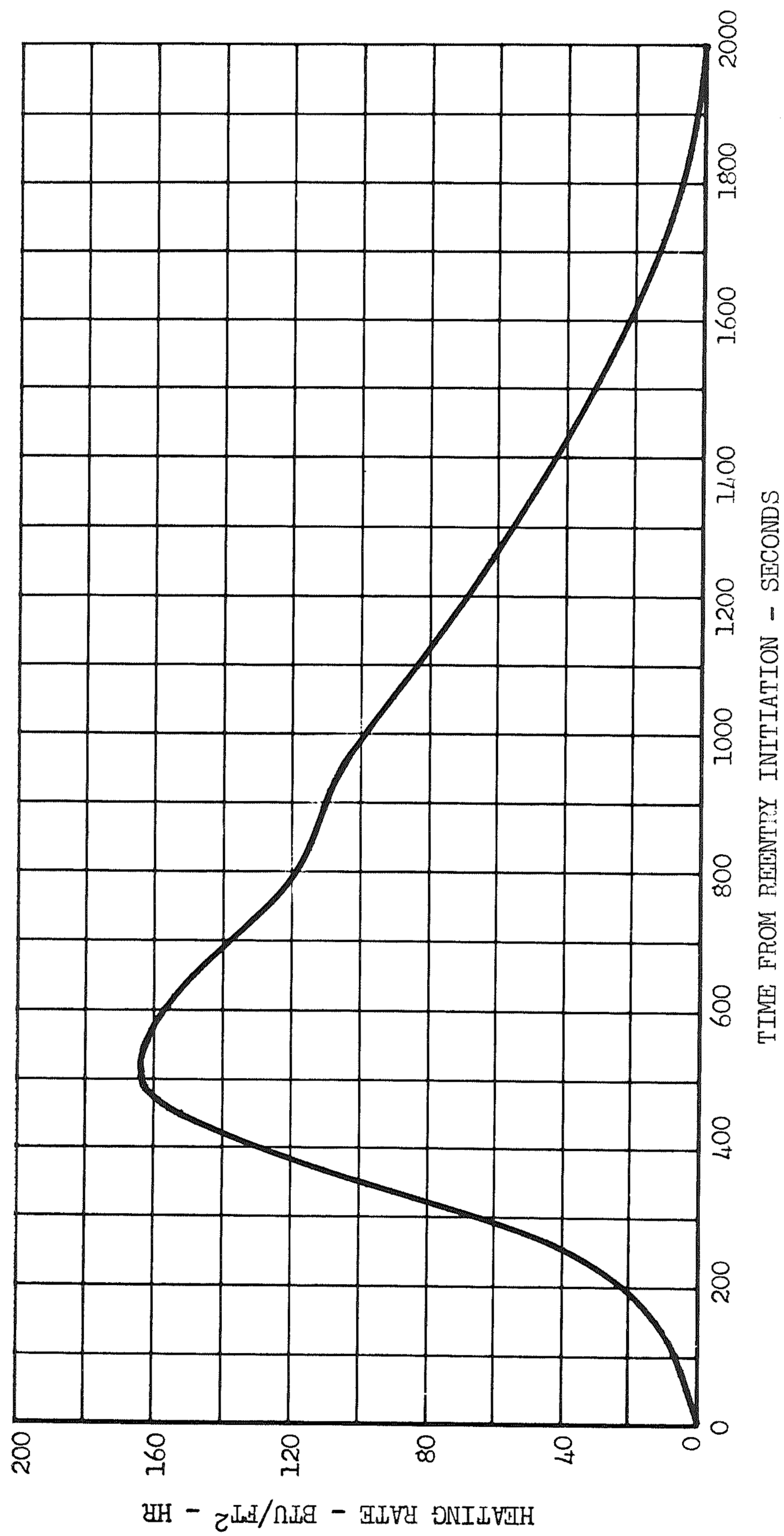
Orbiter reentry thermal histories are presented in Figure A-8 for the inner insulation surface and for the tank wall at upper and lower side and at bottom locations. The difference in thermal histories for the various locations is occasioned by differences in local heating rates and variations in insulation time constants. Reentry analysis demonstrates slow tank and inner insulation surface cooling rate, even after reentry completion. This requires APS components to withstand reentry thermal environment for a much longer time than reentry actually takes. However, natural convection, not included in the analysis, would provide more rapid cooling than that shown.

The increase in heat transfer rates to the main engine tank during reentry is of a particular interest in considering methods of maintaining tank pressure during APS usage. A conservative measure of this increase is provided by evaluating heat flux from the insulation surface to the tank during reentry. A mean value, weighing bottom and side heating according to surface areas exposed, is given in Figure A-9.



LCR TYPICAL REENTRY TEMPERATURES
OF TANK-INSULATION SURFACES

FIGURE A-8

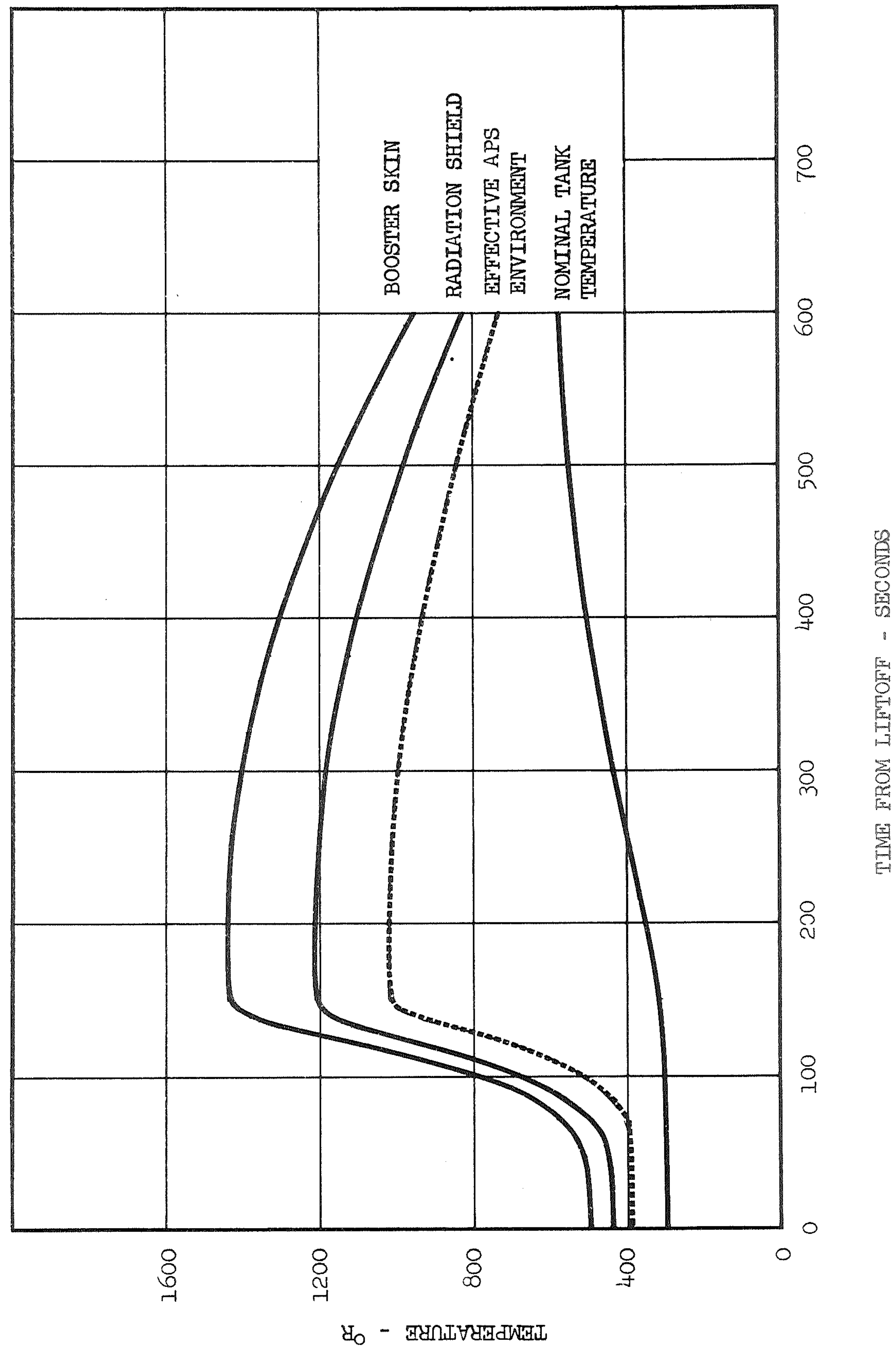


MEAN HEAT FLUX INCREASE TO ORBITER MAIN TANK DURING REENTRY

FIGURE A-9

A-5. BOOSTER ASCENT AND REENTRY HEATING

The ascent thermal environment experienced by the booster is similar to that of the orbiter. However, the booster reentry heating is substantially less. The booster internal thermal environment was based on the effective internal environmental temperature, which was determined by the radiation average of the skin and tank temperatures. These estimates are shown in Figure A-10.



TYPICAL BOOSTER MISSION TEMPERATURE PROFILES

ON TOP OF VEHICLE NEAR CANARD

FIGURE A-10

APPENDIX B

MAIN ENGINE TANK THERMODYNAMICS

B.1 INTRODUCTION

Main engine tank thermodynamic processes are the most important consideration in low pressure APS design. These processes have a direct bearing on the determination of:

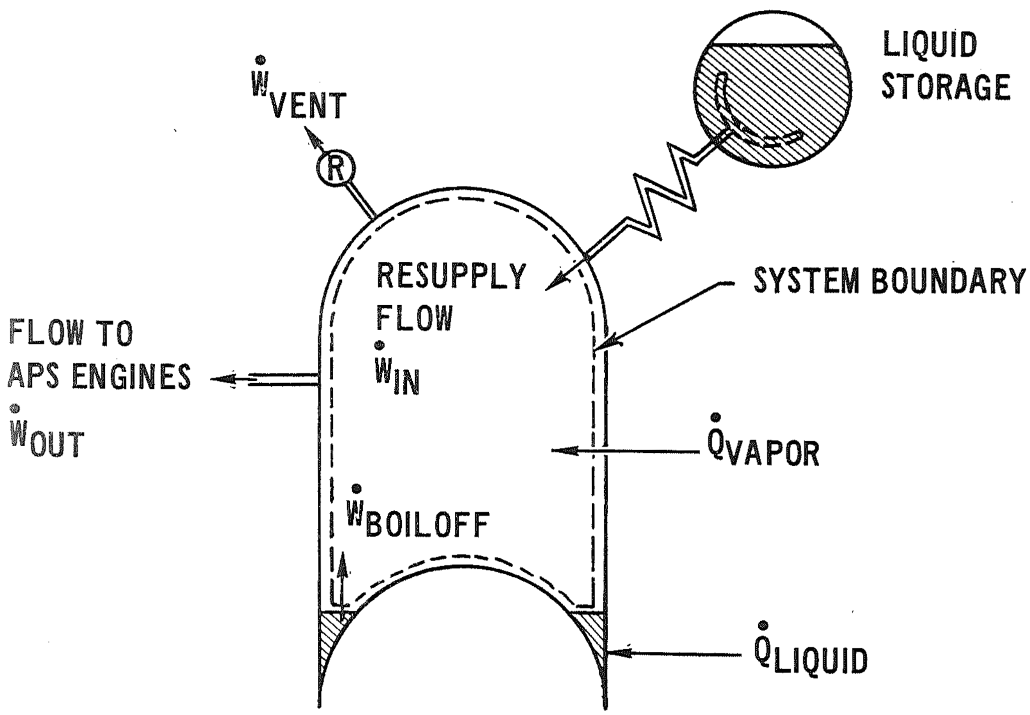
(1) propellant storage requirements, due to the potential utilization of boost residual propellants

(2) propellant distribution network and controls design, due to the impact of main engine tank design pressures on available subsystem pressure budget, and

(3) heat exchanger design, due to the utilization of main engine tank heat capacity for conditioning of resupply propellants.

All APS design and analyses were performed assuming the main engine tank vapor mass was uniformly mixed. This permitted evaluation of main engine tank pressure and temperature fluctuations with varying boundary conditions, applying the classical equations for conservation of energy and mass. It was recognized, however, that because of large tank sizes and low on-orbit vehicle accelerations, complete mixing of tank ullage vapor was unlikely. Therefore, additional studies were undertaken to identify the potential impact of propellant vapor stratification on APS design and the characteristics of different propellant resupply concepts were explored. This appendix provides a description of the analytical approach used for evaluation of main engine tank thermodynamics. Heat transfer considerations, including potential effects of external environmental changes, and mass transfer, including the rationale used to establish resupply and residual control methods, are discussed. Additionally, the potential for and the effects of thermal stratification on APS operation and the general mixing characteristics of alternate resupply/injection concepts are described. A brief description of the analyses conducted to describe main engine tank thermodynamics, from the standpoint of both complete and incomplete mixing, and a discussion of the general influences of main engine tank thermodynamics on APS design are the subject of this appendix.

B.2 Main Engine Tank Thermodynamics (Complete Mixing) - The thermodynamic state of propellant fluids within the main engine tanks is of fundamental importance to the low pressure APS. In general, this is an open, unsteady system as shown by the illustration of Figure B-1. Pressure and temperature changes during a given time interval are a function of initial fluid mass and thermodynamic state, and the



MAIN ENGINE TANK THERMODYNAMIC SYSTEM

DEFINITION

FIGURE B-1

difference between energy entering (i.e., resupply propellant flow and heating rate) and energy leaving (i.e., total engine flow) the system. System temperature change was calculated applying energy conservation, assuming a homogeneous propellant vapor mass. Using this temperature change and the conservation of mass, system pressure change was computed. Performing calculations in this manner, main engine tank pressure and temperature profiles were determined during APS operation.

The boundary conditions associated with the main engine tank thermodynamic system were categorized as:

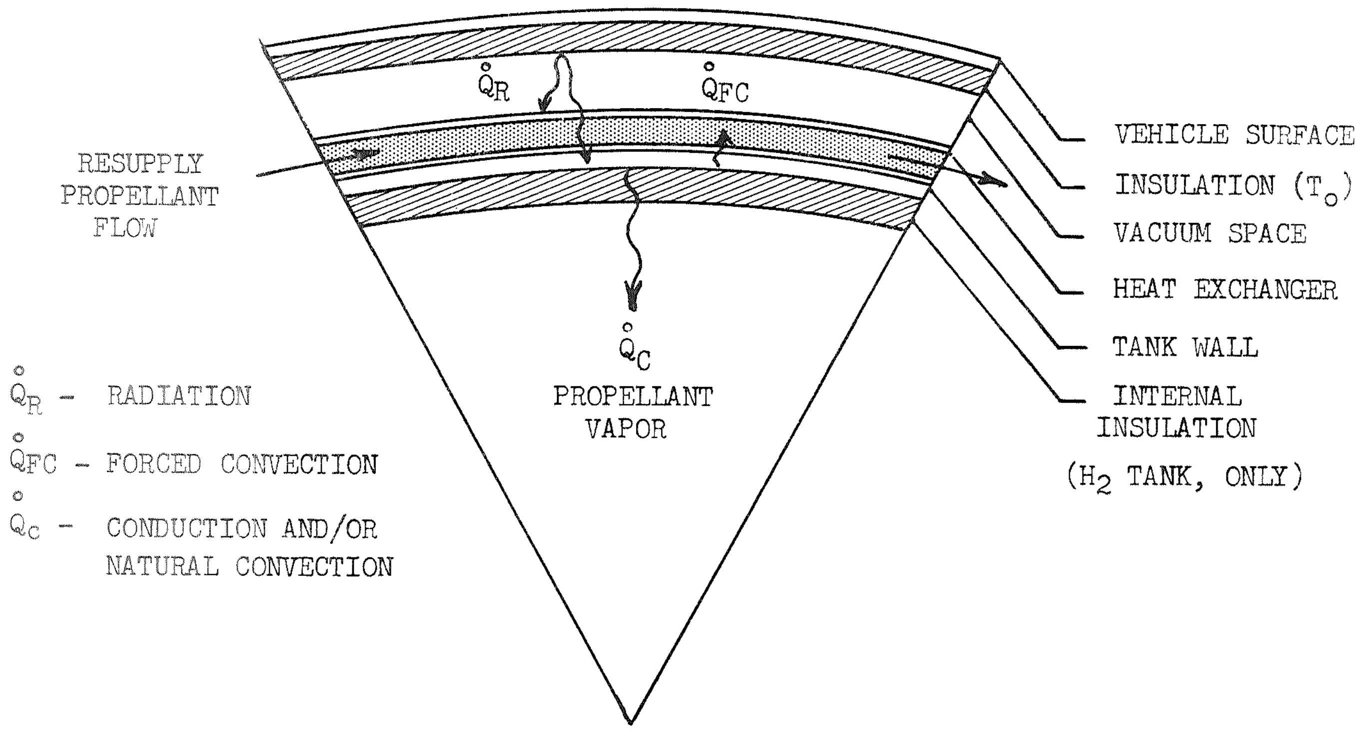
(1) heat transfer including heat transfer to the tank walls and from the walls to the vapor, and

(2) mass transfer, including main engine tank resupply and engine flow, residual liquid propellant boil-off, and tank venting.

Definition of these boundary conditions and discussion of their impact on APS design are provided in the following paragraphs.

B2.1 Heat Transfer - The general model employed for evaluating main engine tank heat transfer rates is illustrated in Figure B-2. Heat is transferred from the surrounding insulation (at temperature T_o) to the tank wall and heat exchanger tubing by radiation, and from the tank wall/tubing to heat exchanger fluids by forced convection. Heat is transferred between the tank wall and internal propellant vapor by the mechanisms of natural convection and/or conduction. Natural convection prevails when vehicle control accelerations cause the convective heat transfer coefficient to be greater than the conductive coefficient. Conversely, when vehicle control accelerations are very low, heat transfer between the tank wall and propellant vapor is by conduction, only. A parametric comparison of heat transfer rates by convection and conduction is shown in Figure B-3 for the orbiter main engine oxygen tank.

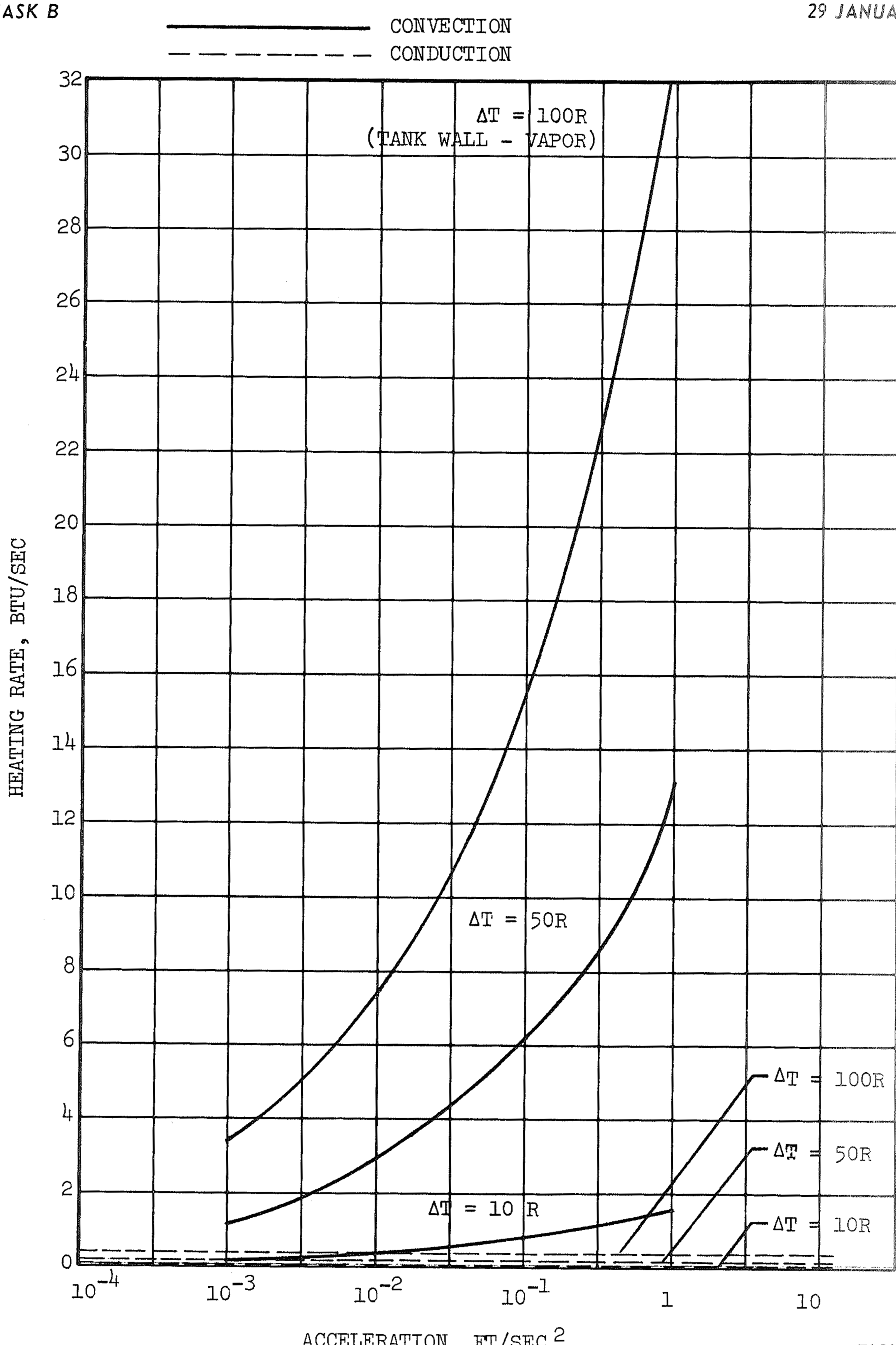
The heating rate mechanisms described above impact APS design by their effect on main engine tank pressure and resupply propellant requirements. If tank-to-vapor heating is reduced (i.e., by decreasing the surrounding environmental temperature, T_o) main engine tank pressure decreases during APS operation. Unless engine feedline diameter is increased to compensate for the reduced pressure-budget, engine flowrate (thrust level) will decay during critical events. An example of this is shown in Figure B-4. Here the critical event occurs at approximately 26 hrs into the mission. Two external temperatures are shown and the increased pressure decay with reduced heating is pronounced. Vapor temperature in the tank is also lowered and less liquid is utilized in the downstream liquid/vapor mixer.



MAIN ENGINE TANK HEAT TRANSFER

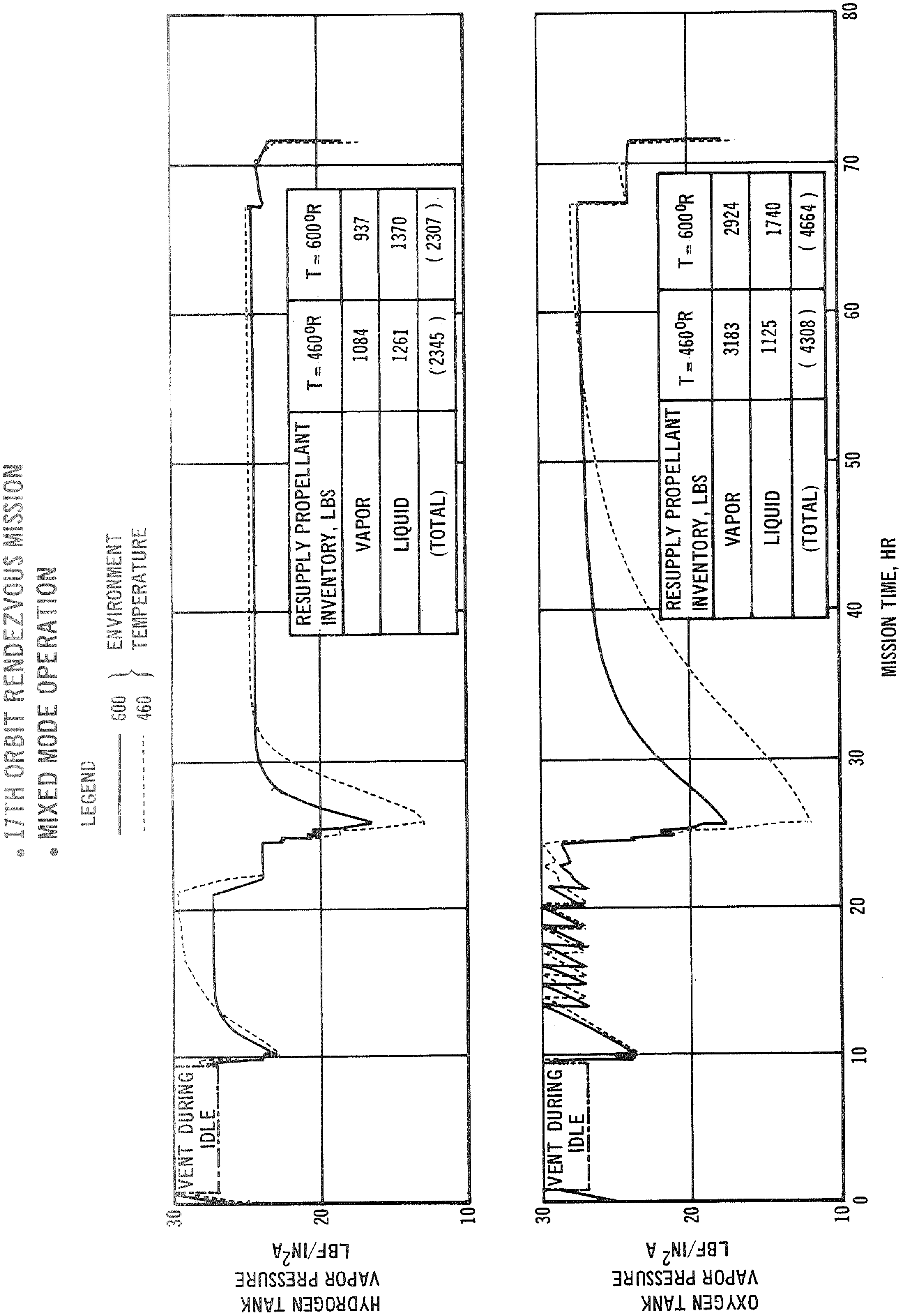
MECHANISMS

FIGURE B-2



MAIN ENGINE O_2 TANK VAPOR HEATING RATES

FIGURE B-3



Hydrogen tank:

RESUPPLY PROPELLANT INVENTORY, LBS	T = 460°R	T = 600°R
VAPOR	1084	937
LIQUID	1261	1370
(TOTAL)	(2345)	(2307)

Oxygen tank:

RESUPPLY PROPELLANT INVENTORY, LBS	T = 460°R	T = 600°R
VAPOR	3183	2924
LIQUID	1125	1740
(TOTAL)	(4308)	(4664)

EFFECT OF ENVIRONMENTAL TEMPERATURE EXTREMES

FIGURE B-4

Consequently, vapor resupply to the main engine tank increases (inset, Figure B-4). Conversely, however, high tank-to-vapor heating rates result in significant venting and consequently greater overall resupply propellant demands. The net effect of lower tank heating is a reduced total resupply propellant requirement due to a reduction in propellant vent losses for the total mission. This effect is illustrated in the example of Figure B-5.

B2.2 Mass Transfer - Propellant flow into and out of the main engine tank vapor system has a direct bearing on propellant feed pressure and resupply propellant demands. Propellant mass is transferred across the system boundary by engine flow, resupply flow, residual liquid boil-off, and venting.

Resupply/Engine Propellant Flow - If engine propellant flow demands are high, main engine tank pressures could decay to unacceptable levels for control of engine thrust. To preclude this effect, energy is added to the main engine tanks in the form of resupply propellant flow. The resupply propellant energy level is a function of both mass flow rate and enthalpy (temperature). An example of how these parameters affect main engine tank vapor pressure and temperature is shown in Figure B-6. In this example resupply propellant temperature is plotted as a function of the ratio of resupply to engine flow rate. Shown are resupply conditions required to: (1) maintain a constant tank vapor pressure; and (2) maintain a constant tank vapor temperature. The curves for constant pressure and temperature coincide at a resupply to engine flow ratio of unity and a resupply propellant temperature equal to the initial tank vapor temperature. At higher resupply temperatures, there is an increase in both vapor pressure and temperature. Conversely, at lower resupply temperatures, tank vapor pressures and temperatures decrease. Studies presented in Reference (a) established that a resupply to engine mass flow ratio of unity was optimum, since it minimized total propellant weight. The remaining problem, then, was to establish resupply propellant temperatures for the critical mission events to meet minimum required engine thrust levels. This result, in turn, established the required heat exchanger size (flow and surface areas).

Two switching techniques were investigated for initiating resupply of the main engine tanks:

(1) a simple pressure sensing technique; and

(2) a vapor mass sensing scheme.

In the first approach main engine tank pressure was allowed to decay during maneuver burns or limit cycle operation until a pressure of 24 lb/in^2a was attained. At

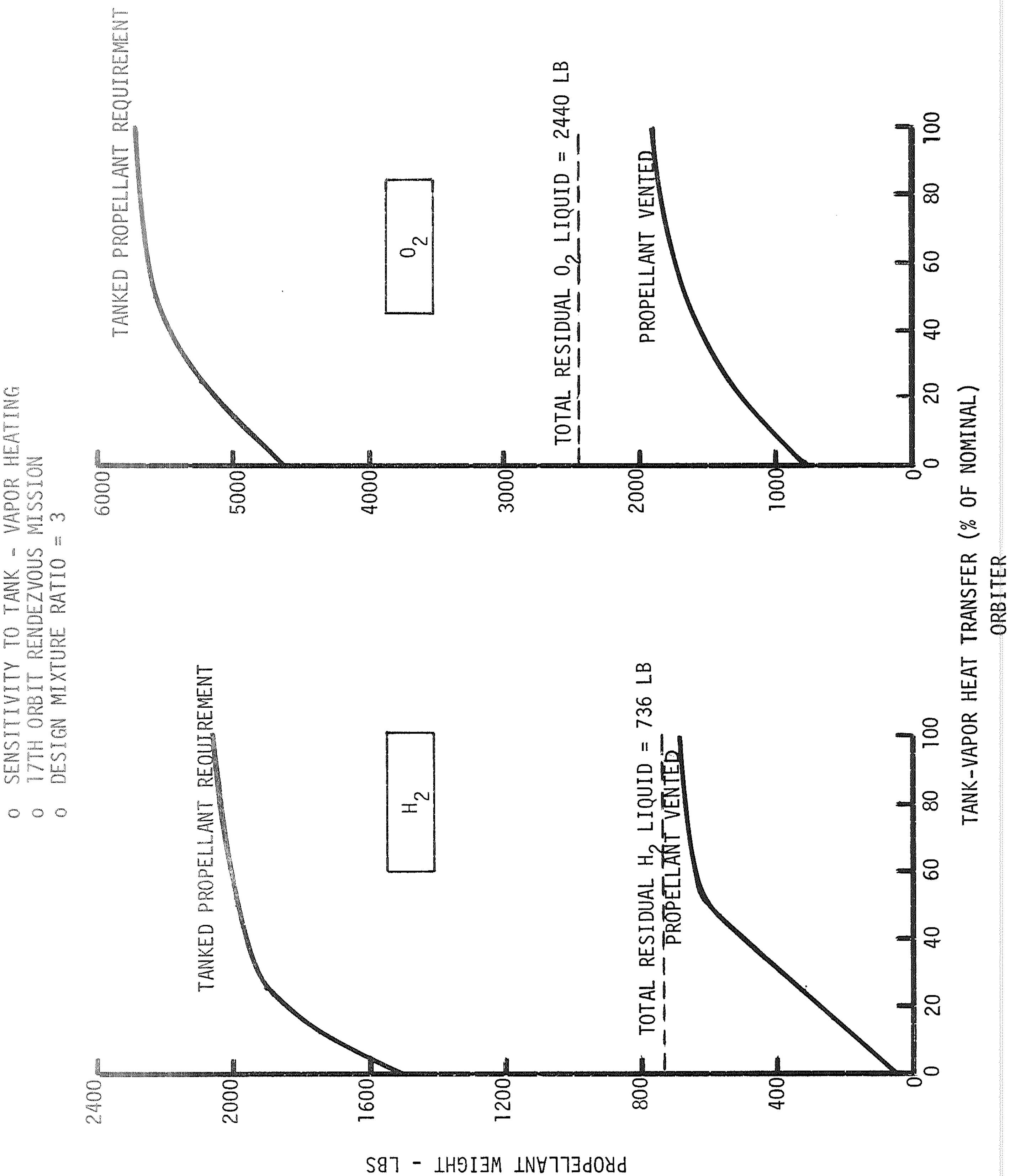
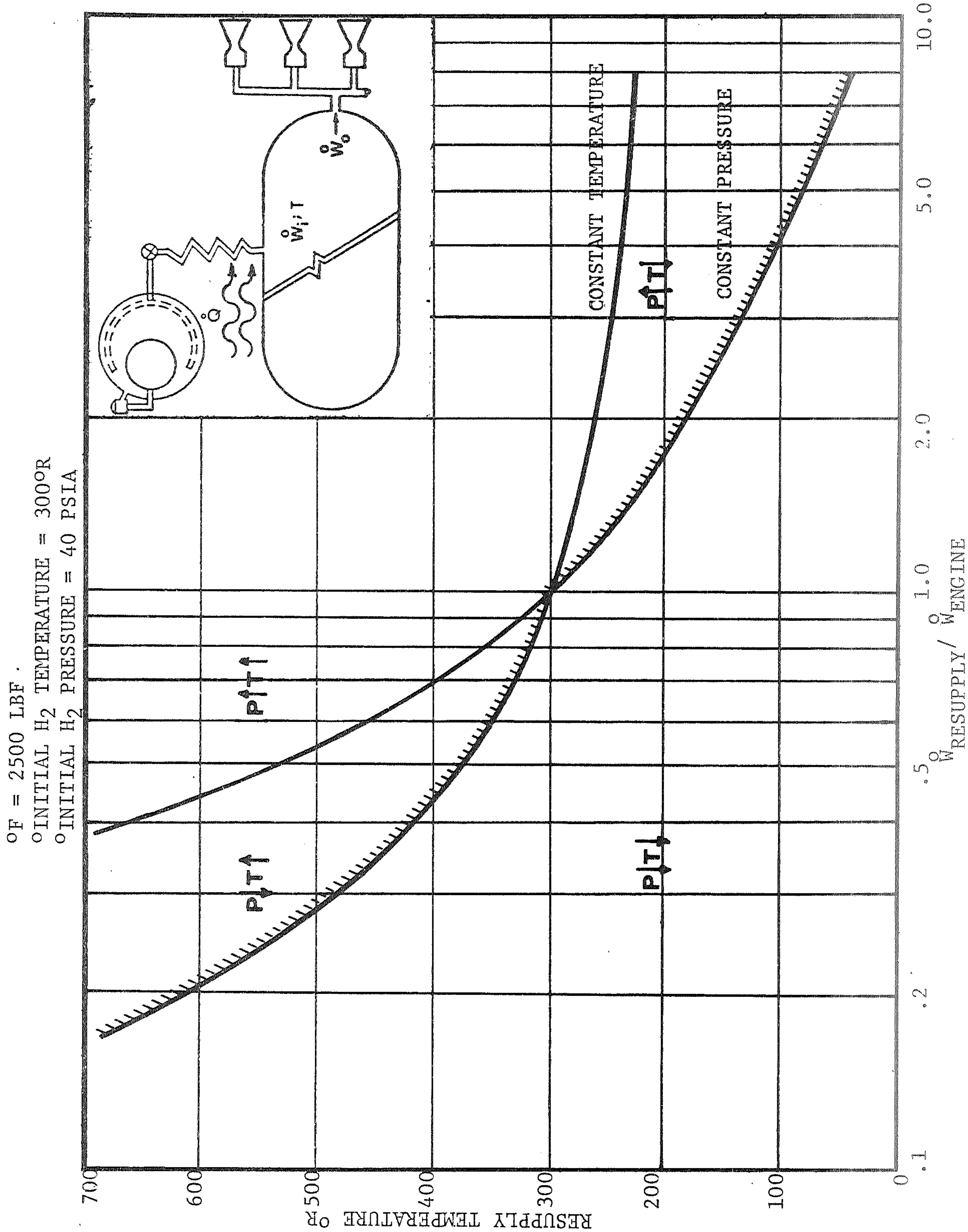

o SENSITIVITY TO TANK - VAPOR HEATING
o 17TH ORBIT RENDEZVOUS MISSION
o DESIGN MIXTURE RATIO = 3
H2
TANKED PROPELLANT REQUIREMENT
TOTAL RESIDUAL H2 LIQUID = 736 LB
PROPELLANT VENTED
O2
TANKED PROPELLANT REQUIREMENT
TOTAL RESIDUAL O2 LIQUID = 2440 LB
PROPELLANT VENTED
PROPELLANT WEIGHT - LBS
0
400
800
1200
1600
2000
2400
0
1000
2000
3000
4000
5000
6000
0
20
40
60
80
100
0
20
40
60
80
100
TANK-VAPOR HEAT TRANSFER (% OF NOMINAL)
ORBITER

FIGURE B-5

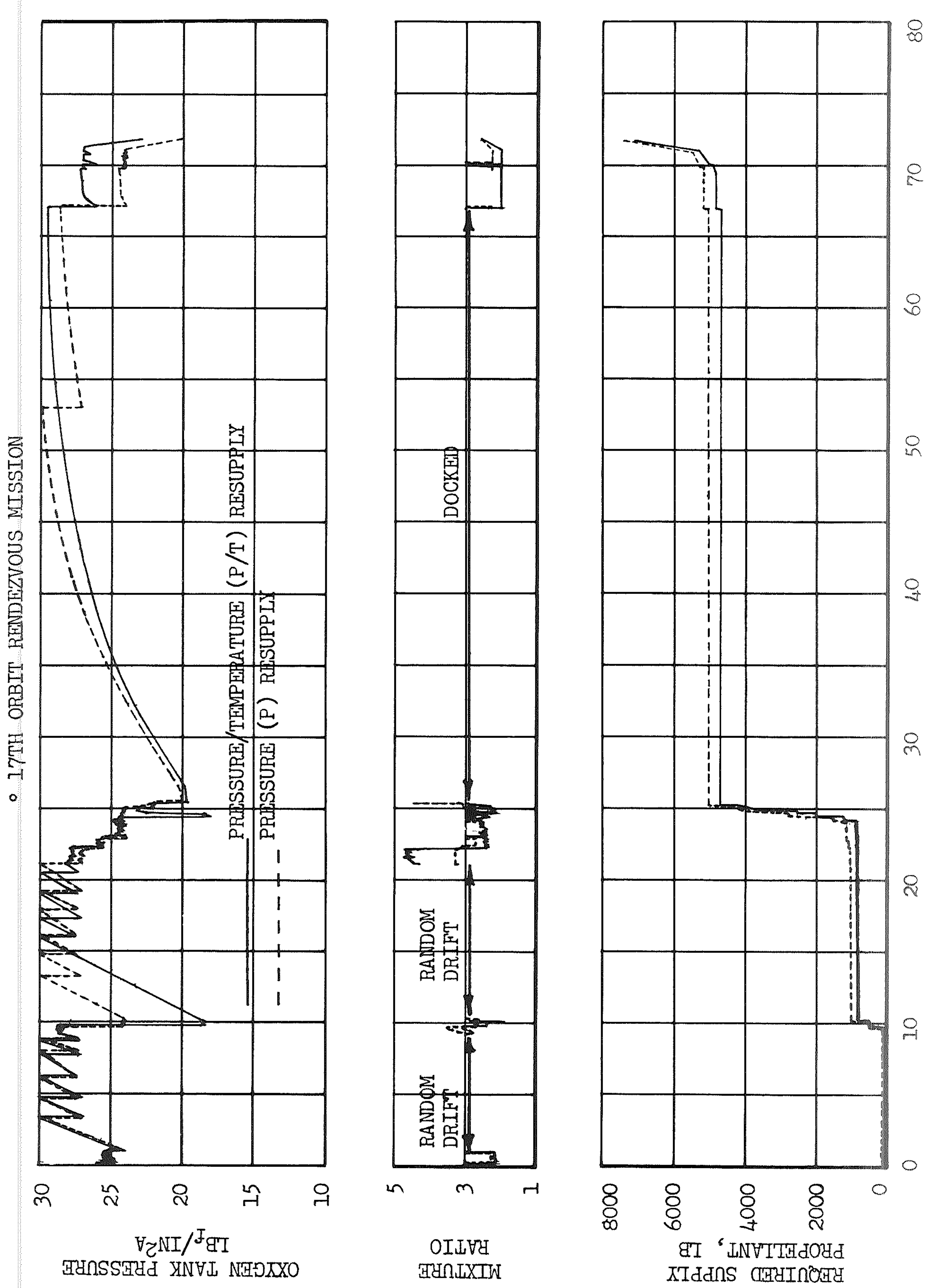


HYDROGEN ASCFNT TANK PRESSURE - TEMPERATURE CHARACTERISTICS

FIGURE B-6

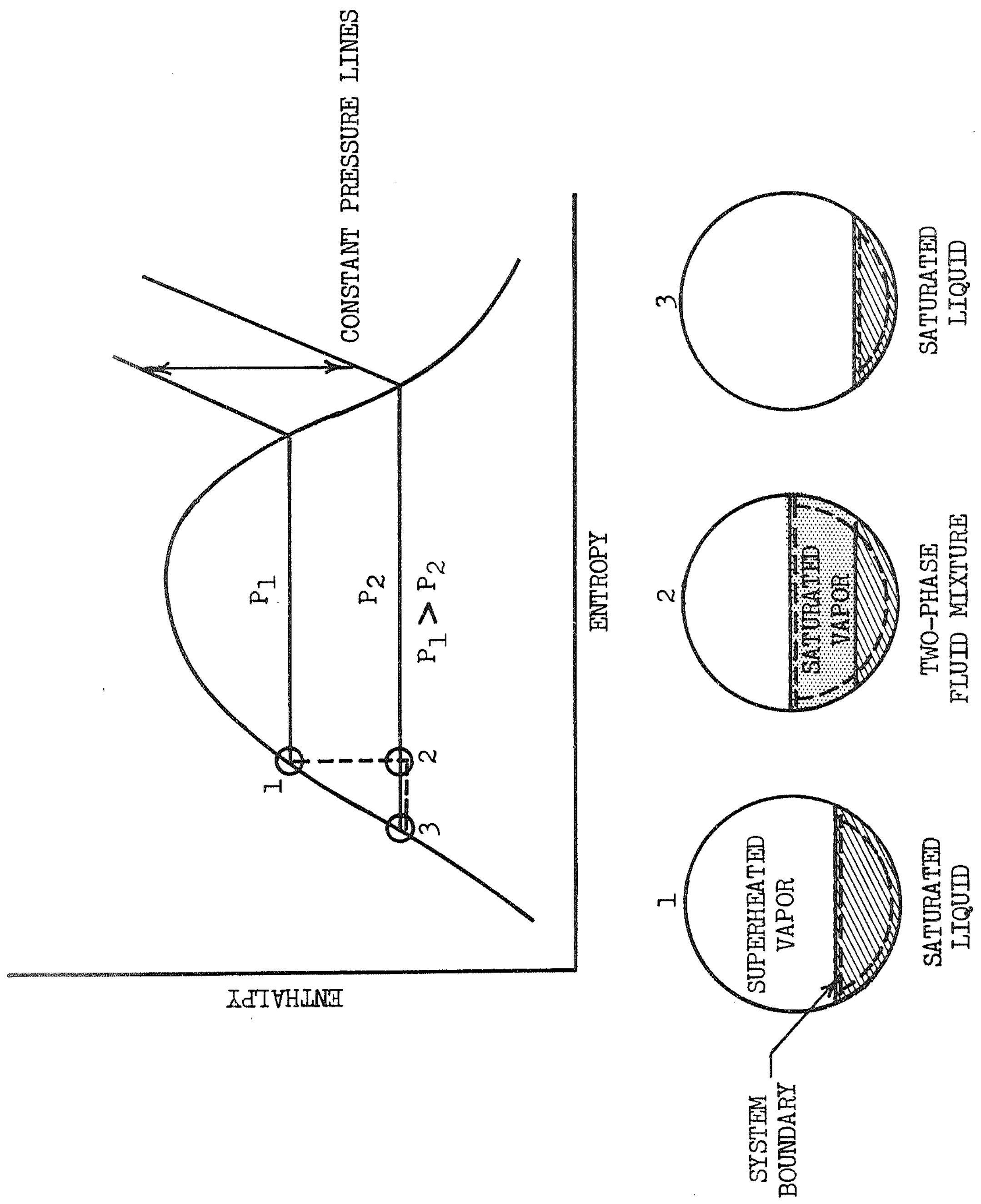
this point resupply propellant flow was initiated in response to an electrical signal from a pressure switch. This supply flow was maintained until completion of the event or until tank pressure was increased to 25 lb/in^2a. In the second approach, main engine tank resupply was initiated in response to a prescribed ratio of pressure and temperature. This was equivalent to initiating resupply after tank vapor mass had decayed to a prescribed value. A pressure/temperature ratio of .0565 (30 lb/in^2a tank vent pressure/530°R tank external environment temperature) was employed. Unlike the first approach, which was based on resupply only during APS usage, main engine tank resupply was allowed during idle periods as well as during maneuver burns and limit cycle periods. Comparison of mission pressure and resupply propellant histories for the two switching approaches is shown in Figure B-7. As seen from this comparison, total propellant resupply requirements were reduced by 187 lbs using the P/T switching technique. Because of this propellant weight savings, the P/T switching technique was selected for initiation of main engine tank resupply.

Residual Liquid Boil-off and Venting - At the time of orbiter main engine shutdown, there are significant quantities of oxygen and hydrogen liquid residuals remaining in the orbiter and booster main engine tanks. These residual propellants are assessed entirely against the main propulsion subsystem, and, if a portion of these propellants can be utilized by the APS to satisfy impulse requirements, APS weight can be reduced. Determination of the amount of residuals that can be credited to the APS requires an analysis coupling the effects of heat transfer into the main engine tanks, residual liquid vaporization, and APS usage. The greatest uncertainty in this analysis is the heat transfer and vaporization rates associated with tank residual liquids. A model, developed to predict residual liquid heating and corresponding vaporization rates is discussed in Appendix C of Reference (a). Corrections to this model were made during Subtask B studies to account for flash boiling which occurs whenever the ullage pressure drops below the residual liquid saturation pressure. In order to account for this effect, the model illustrated in the H-S diagram of Figure B-8 was employed. As shown, saturated liquid is assumed at point (1). As tank pressure decays to point (2) the residual fluid quality is increased from zero to a value, x. The new quality is readily calculated from a knowledge of saturated entropies at pressures corresponding to points (1) and (2). Saturated vapor then flashes off the residual fluid and mixes with the warmer ullage vapor, returning the liquid to a saturated state at point (3). Simulated APS mission duty cycles were conducted for both the booster and orbiter using this



COMPARISON OF PRESSURE CONTROL CONCEPTS
INCLUDING RESIDUALS EFFECTS
(NON-COMPARTMENTED TANKAGE)

FIGURE B-7



LIQUID FLASH BOILING MODEL

FIGURE B-8

flash-boiling model and the results are shown in Figure B-9 and B-10, respectively. Most importantly, it is seen that total booster impulse requirements can be satisfied entirely using residual propellants contained in the main engine tanks following boost. For the orbiter, however, only 19 percent of residual liquids could be utilized by the APS for the main engine tank configuration specified in the Space Shuttle Vehicle Description and Requirements Document (SSVDRD). This relatively poor residual liquid utilization resulted from high vaporization rates which caused most of the liquid boil-off to be vented before major vehicle velocity changes were performed.

Since liquid oxygen constituted the greatest residual mass, alternate schemes were considered for obtaining better utilization of oxygen residuals. The preferred approach is illustrated in Figure B-11. It employs a separate compartment between bulkheads in the main engine oxygen tanks. Liquid oxygen residuals would be trapped in this compartment following boost. Subsequent environmental heating of the residual oxygen causes a more rapid pressure rise in the compartmented tank than main engine tank due to the larger ratios of wetted tank area-to-propellant mass. When a pressure differential of 4 lb/in^2 (maximum bulkhead pressure differential) is attained, a relief valve between the main oxygen and compartmented tank opens and allows flow between the tanks until the pressures equalize. In this manner, passive control is exercised over liquid oxygen boil-off, and as a result greater utilization of residual oxygen is achieved. This is illustrated by the example of Figure B-12, which compares mission main engine and compartmented tank pressure profiles and tabulates residual liquid oxygen utilization. As seen, an additional 1064 lbs of residual oxygen is used by the APS. Based on this substantial weight advantage, the compartmented O_2 tank was selected as a baseline design for the orbiter APS.

B-3 <u>Non-Homogeneous Heating/Mixing</u> - Investigations were conducted to determine the potential deviation in main engine tank temperature from a homogeneous model during orbiter mission phases, and to determine corresponding effects on APS design. Studies of orbiter mission phases determined that substantial vapor stratification can occur in the main engine tanks and that the type and location of main tank injection/extraction ports can have a significant effect on tank pressure variation and APS propellant requirements. These effects are discussed in the following paragraphs.

<u>Propellant Vapor Stratification</u> - The potential for main engine tank propellant vapor stratification was evaluated using the model shown in Figure B-13. The

	H_2	O_2
RESIDUALS		
LIQUID	6060	14,285
VAPOR	826	4,511
TOTAL	(6886)	(18,796)
APS PROPELLANT		
RESIDUAL UTILIZATION	875	1,769
RESUPPLY	0	0
TOTAL	(875)	(1,769)

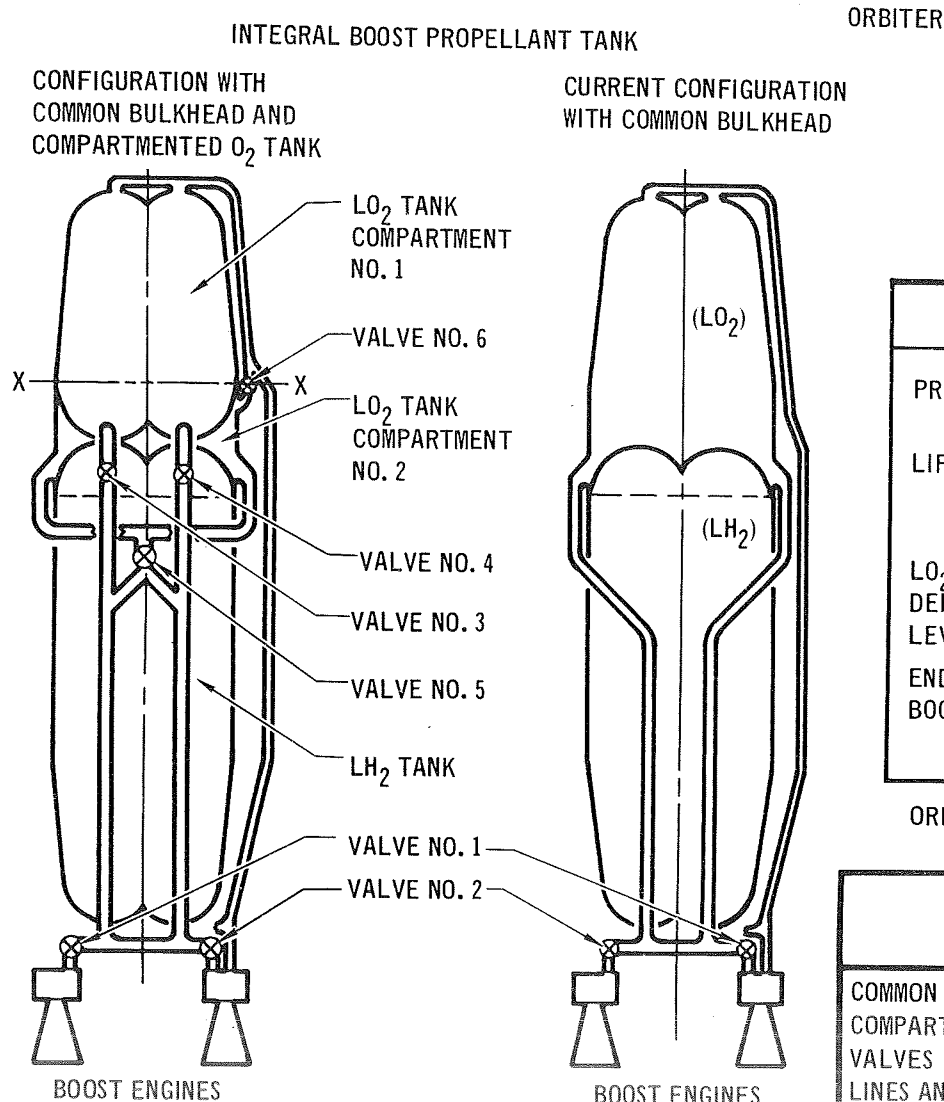
BOOSTER RESIDUAL USAGE

FIGURE B-9

	H_2	O_2
RESIDUALS		
LIQUID	736	2,440
VAPOR	167	1,191
TOTAL	(903)	(3,631)
APS PROPELLANT		
RESIDUAL UTILIZATION	0	1913
RESUPPLY	2499	4496
TOTAL	(2499)	(6409)

ORBITER RESIDUAL USAGE

FIGURE B-10



SEQUENCE OF OPERATION FOR CONFIGURATION WITH COMPARTMENTED O_2 TANK

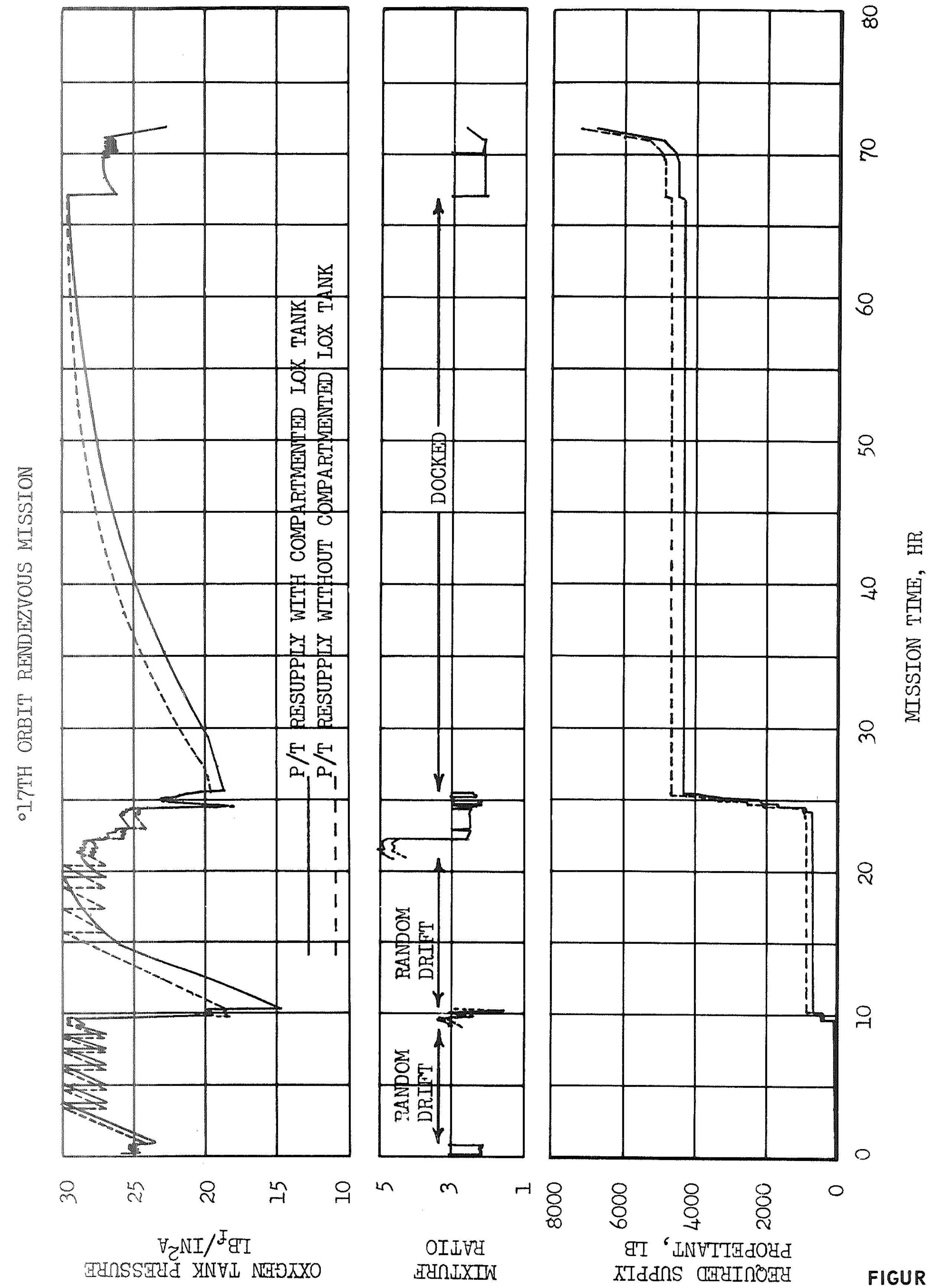
EVENT	VALVE OPERATION	REMARKS
PRELAUNCH	NOS. 1 & 2 CLOSED NOS. 3, 4, 5, 6 OPEN	LOAD COMPARTMENTS NOS. 1 & 2
LIFTOFF	ALL VALVES OPEN EXCEPT NO. 5	ALL LO_2 FLOW FROM COMPARTMENT NO. 1
LO_2 SUPPLY DEPLETED TO LEVEL X--X	OPEN NO. 5	LO_2 FLOW FROM COMPARTMENTS NO. 1 & 2
END OF BOOST	CLOSE ALL VALVES EXCEPT NO. 5	ALL O_2 RESIDUALS ARE CONTAINED IN COMPARTMENT NO. 2.

ORBITER TANK WEIGHT COMPARISON

ITEM	BOOST TANK CONFIGURATION: COMMON BULKHEAD	BOOST TANK CONFIGURATION: COMPARTMENTED O_2 TANK
COMMON BULKHEAD	950 LB	540 LB
COMPARTMENT BULKHEAD	–	210 LB
VALVES	–	175 LB
LINES AND MOTION COMPENSATORS	–	75 LB
	950 LB	1000 LB

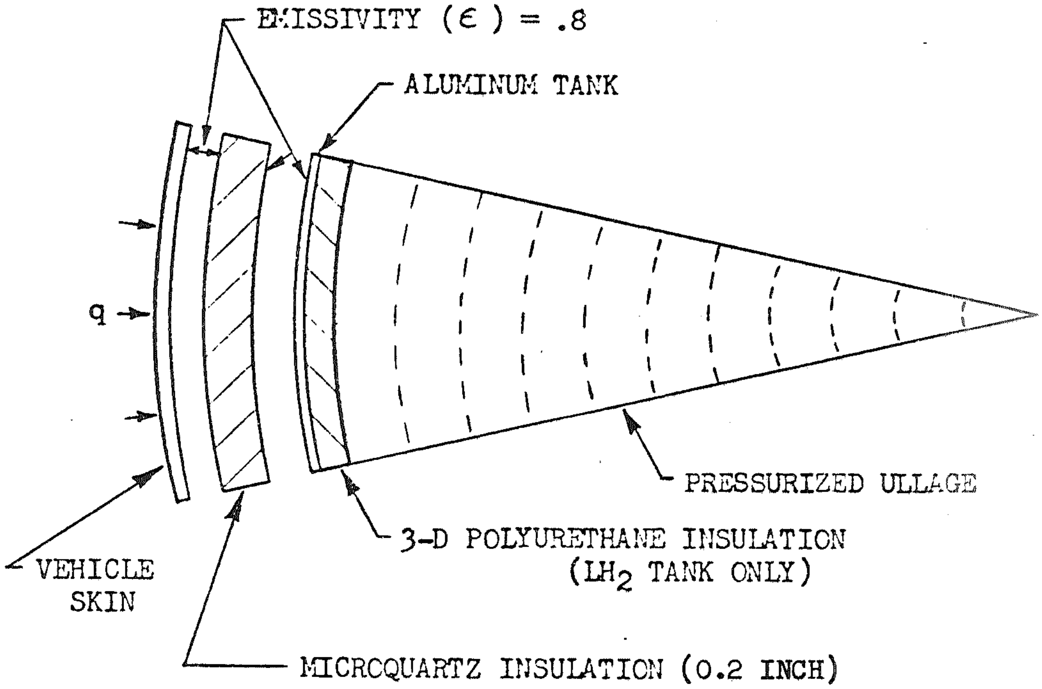
COMPARTMENTED OXYGEN PROPELLANT TANK

FIGURE B-11



EFFECT OF COMPARTMENTED TANK
ON APS PERFORMANCE

FIGURE B-12



THERMAL MODEL

FIGURE B-13

micro-quartz insulation thickness is typical of a location on the side of the orbiter and the heat flux was selected to provide an equilibrium radiation temperature of 500°R. The gas model consists of 11 radial nodes with an initial uniform density corresponding to a pressure of 18.5 psia and a specified temperature. Step temperature changes between the tank wall and vapors were assumed, typifying the non-equilibrium conditions which can exist following an APS maneuver.

Temperature profiles from these analyses are shown in Figure B-14 for the hydrogen and oxygen tanks, respectively. Substantial heating of the vapor near the wall occurs almost immediately due to the internal energy (mass and heat capacity) of the tank wall. The oxygen vapor temperature gradients are substantially larger and decay more slowly than the hydrogen. This is caused by the low oxygen thermal conductivity. With increasing time, temperatures and pressures asympotically approach equilibrium values. These results do not account for local heat sources, sinks nor natural convection established during the burn but are believed to be fairly representative of the non-uniform conditions to which the APS must operate.

Utilizing these results, a study was performed to evaluate the effect of extracted vapor temperature on APS performance. Three options were evaluated for an environmental temperature of 530°R and an initial bulk vapor temperature of 516°R. The selected options were:

(a) constant low temperature vapor extraction at 350°R

(b) extraction from homogeneous mixture (Baseline case)

(c) constant high temperature vapor extraction at 550°R

Main engine tank pressure profiles for each option are presented in Figure B-15 for the critical pre-docking maneuvers. As shown, tank pressure variation is minimized when cool vapors are extracted (due to the low specific volume of the extracted vapors). However, this results in higher total mass flow through the heat exchangers and main engine tank assemblies. Although the extraction temperature must eventually equilibrate to the bulk mass temperature, the data of Figure B-15, demonstrates the desirability of extracting cool vapors to maintain high tank pressure levels and a potential for improved APS design.

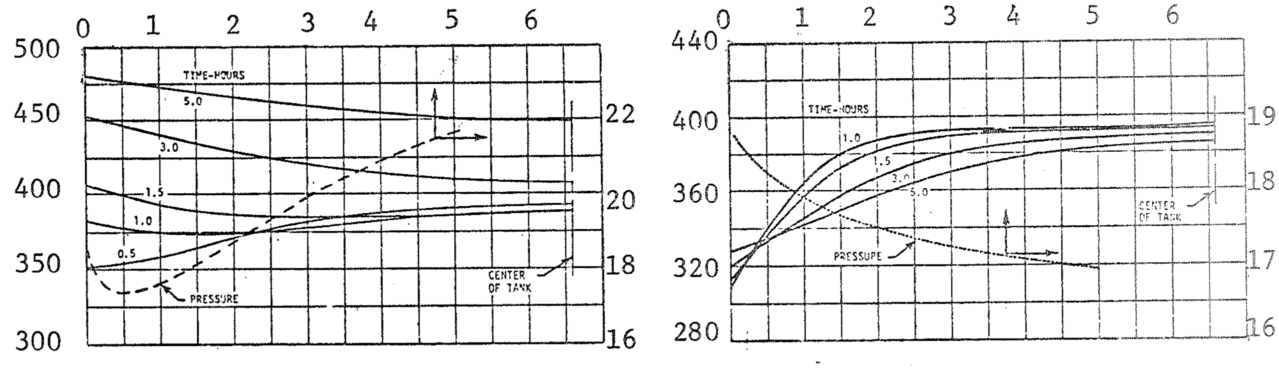
During major maneuver burns and reentry, the temperature profiles shown in Figure B-14 are reoriented from a radial to a longitudinal distribution due to natural convection under imposed g loads. To investigate this phenomenon, a simple physical model was employed to simulate tank vapor conditions. The tank vapor was assumed to be an infinite gaseous medium initially at rest. The radial tank temperature distribution was approximated by a step temperature change at the tank

HYDROGEN — OXYGEN

$\Delta T = -100°R$

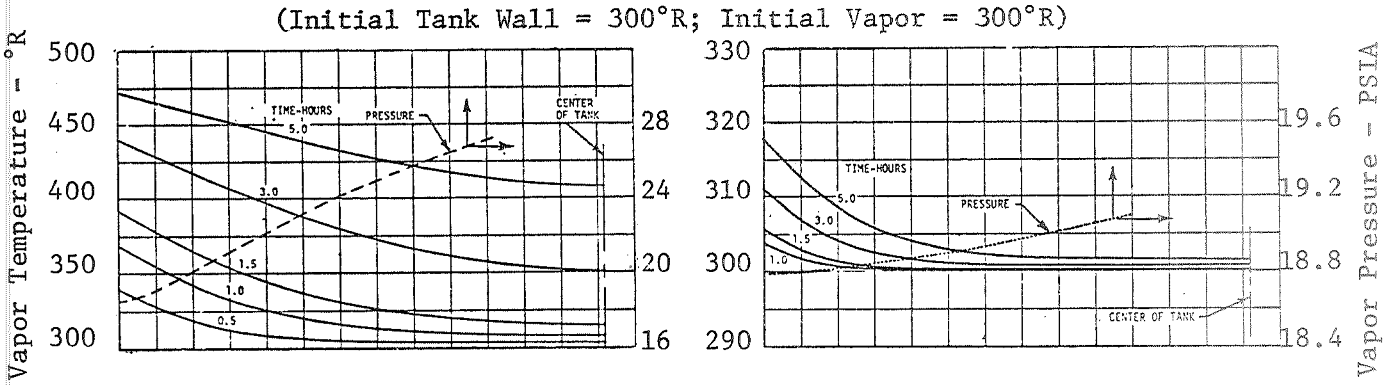
(Initial Tank Wall = 300°R; Initial Vapor = 400°R)

Time for Pressure History, Hours — Time for Pressure History, Hours



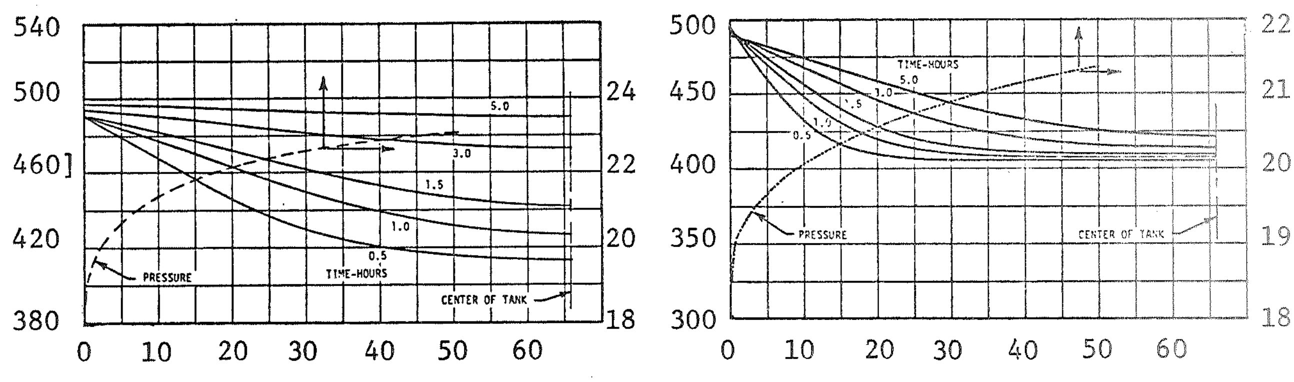
$\Delta T = 0$

(Initial Tank Wall = 300°R; Initial Vapor = 300°R)



$\Delta T = 100°R$

(Initial Tank Wall = 500°R; Initial Vapor - 400°R)



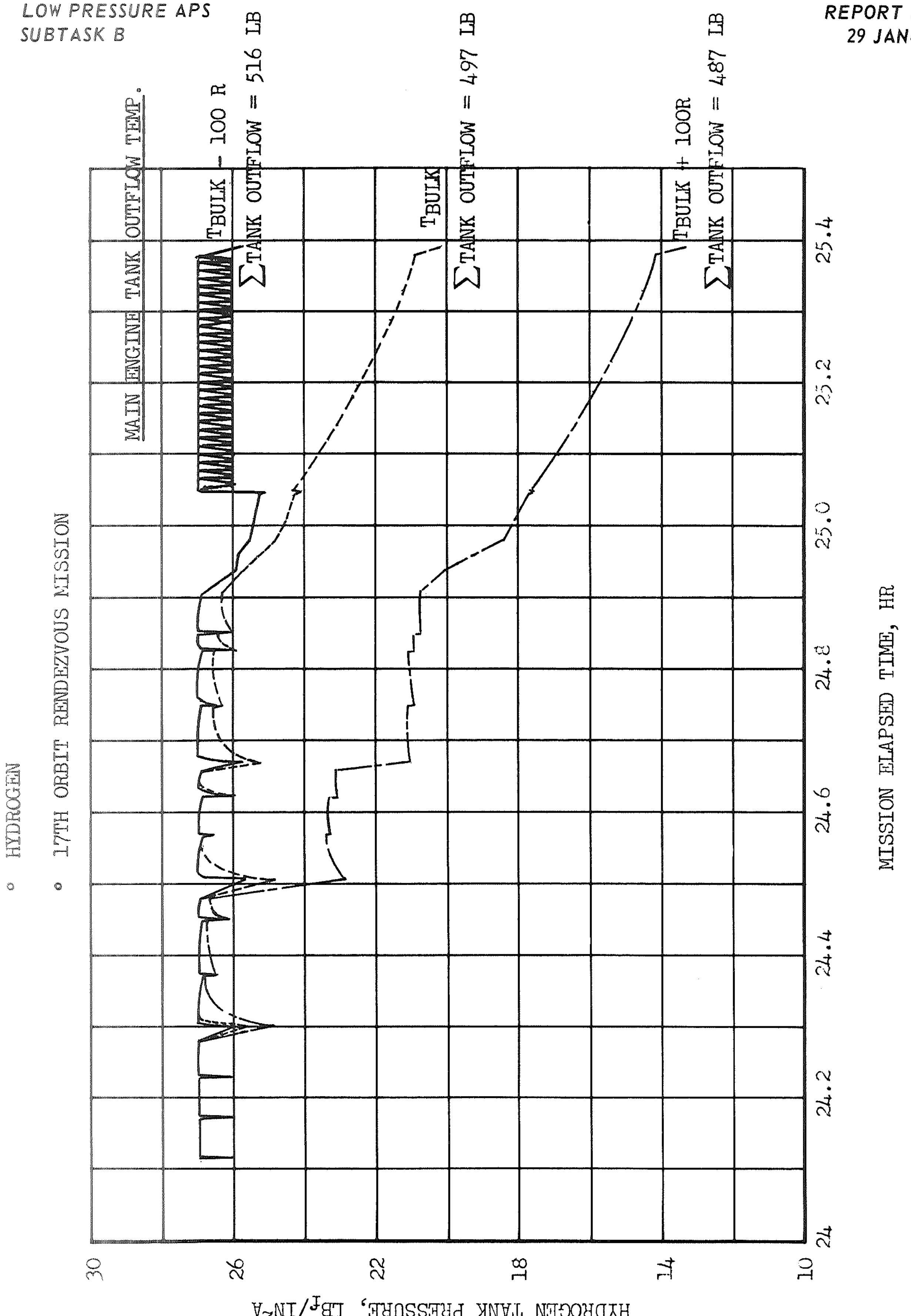
Distance From Tank Wall, Inches — Distance From Tank Wall, Inches

TANK VAPOR PRESSURE AND TEMPERATURE PROFILES

FOR STEADY STATE HEATING

ΔT = (INITIAL TANK WALL TEMP. - INITIAL VAPOR TEMP.)

FIGURE B-14



SELECTIVE MAIN TANK TAP-OFF

FIGURE B-15

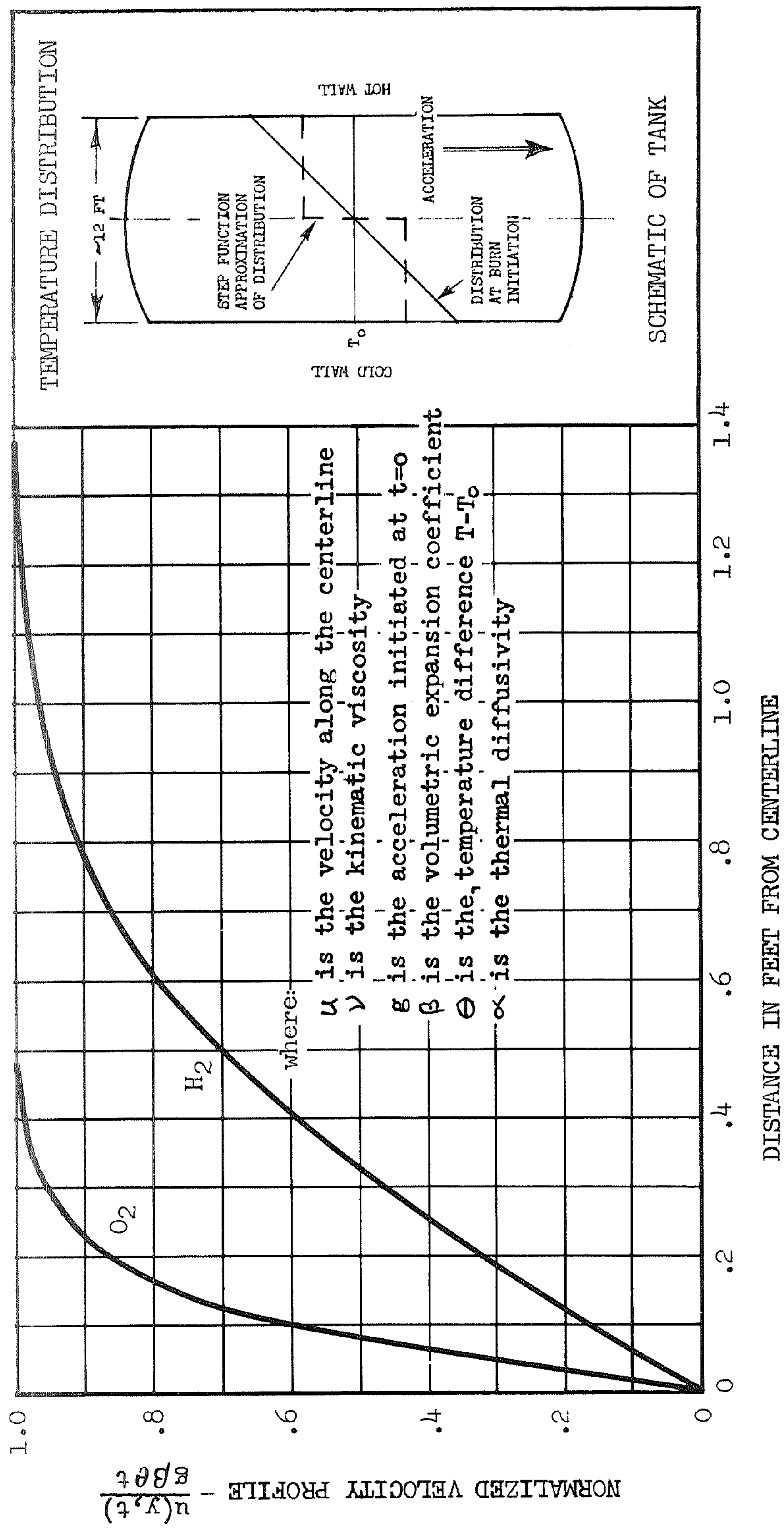
centerline corresponding to the total temperature drop across the tank. Applying the analysis of Reference B, non-dimensional velocity profiles were developed for suddenly accelerated vapor. This is shown in Figure B-16. These profiles show the existence of a convective boundary layer adjacent to the tank centerline during a braking maneuver (velocity imparted from top to bottom in schematic illustration). The warm vapors to the right of center move down whereas vapors to the left of center move up. Vapor velocity at time, t, after start of the maneuver, is given by $U = g\beta\Theta t$ (symbols defined in Figure B-16). Applying this equation to a reasonable range of maneuver acceleration levels, the following approximate vapor velocities were computed:

vehicle acceleration, ft/sec^2	vapor bulk velocity, ft/sec
0.1	0.01(t)
1.0	0.1(t)

Results are equally valid for oxygen and hydrogen. Using a velocity threshold of 3 ft/sec as that required for significant redistribution of vapors, tank-vapor heating is primarily by conduction during the first 30 ft/sec of an APS maneuver; thereafter, vapor heating/mixing is convective in nature. In the example shown, the cooler vapors migrate to the top of the tank and warmer vapors collect near the bottom, thus producing an axial temperature distribution in the direction of the acceleration vector. Cool propellant vapor can then be extracted from the top of the tank to maintain tank pressure.

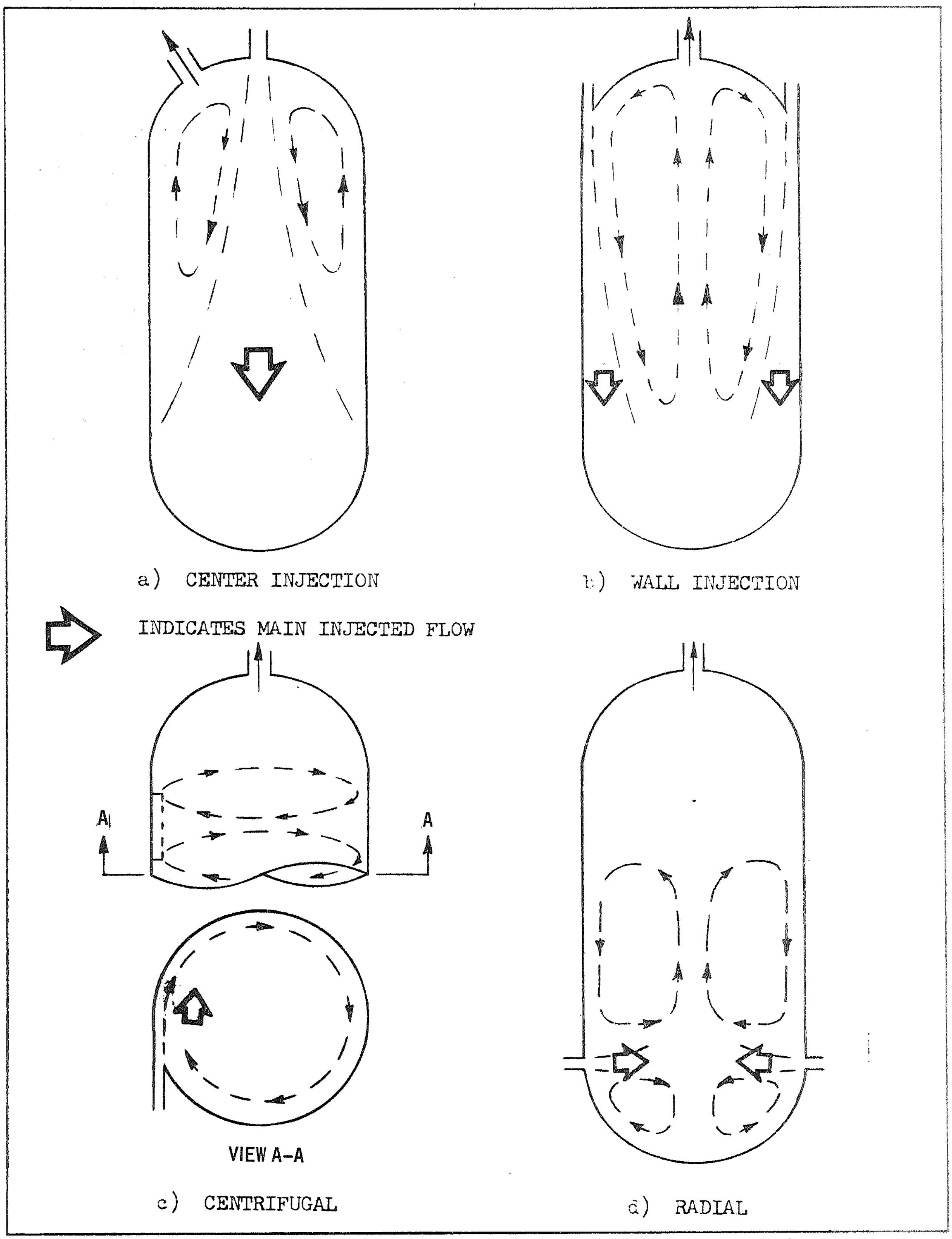
Resupply Propellant Injection - The above examples for orbiter idle and maneuver periods demonstrate the importance of vapor mixing processes. This mixing will be strongly affected by the method chosen for injection of resupply propellant. Alternate resupply injection techniques are shown schematically in Figure B-17. The large arrows indicate the general flow direction while the small arrows on the closed curves illustrate the various secondary flow patterns which would be established for the different schemes. The tank centerline injection scheme offers the simplicity of single port injection and provides minimal thermal communication between the wall and the injected fluid but, if the injected fluid is colder than the tank temperature, this relative isolation is a disadvantage since the thermal energy of the tank wall can not be utilized directly.

The wall injection mode provides a better means of utilizing heat transfer from the wall to the injected flow. Heat transfer from the wall to the resupply flow supplements the heat exchanger and thus aides in maintaining tank pressure during a burn. The difficulty with both center and wall injection modes is that



VELOCITY PROFILE FOR SUDDENLY ACCELERATED VAPOR
WITH INITIAL THERMAL GRADIENT: t = 100 SECONDS

FIGURE B-16



ALTERNATE RESUPPLY INJECTION SCHEMES

FIGURE B–17

they produce significant mixing, thus precluding preferential extraction of cold vapors.

The centrifugal and radial injection modes offer better containment of resupply fluid in the injection near-field, thus establishing and/or maintaining vapor stratification. Radial injection minimizes initial wall contact, but introduces substantial secondary mixing.

The centrifugal injection technique combines the advantage of the wall injection mode with near-field mixing. In this manner wall-to-fluid heat transfer is used to supplement the heat exchanger, and propellant stratification is maintained for selective vapor extraction. Experimental data and analytical models for circumferential wall jets provide means for estimating required lengths and heat transfer processes. Because of these benefits, the centrifugal injection method is preferred.

REFERENCES

A Kendall, A. S., McKee, H. B., Orton, G. F., "Low Pressure Auxiliary Propulsion Subsystem Definition", Subtask A Report, McDonnell Douglas Report No. MDC E0303, dated 29 January 1971.

B Siegel, R., "Trans ASME 80, 347 (1958) Quoted by A. J. Ede in Advances in Heat Transfer, Vol. 4, (1967) p34.

C Myers, G. E., Schauer, J. J., Eustis, R. H., Heat Transfer to Plane Turbulent Wall Jets, Trans. ASME (C), 1963, p. 209-214.

APPENDIX C

C-1. RELIABILITY

Analyses were conducted to ensure that the APS could with a minimum of complexity satisfy fail operational/fail safe requirements of Reference a, and to verify that the basic APS design achieved high reliability. Fail operational/fail safe capability evaluation was performed with functional flow diagrams and preliminary failure mode and effects analyses. These studies defined amount and type of component and subassembly redundancy required, and also constituted the basis for development of Subtask B APS schematics and weights. Component and subassembly failure rates, based on representative, available, failure rate data, were then used to estimate APS reliability. These studies have shown that the low pressure APS design provides a basic simplicity offering potentially high reliability for both booster and orbiter applications.

The general philosophy for implementing fail operational/fail safe redundancy for the Low Pressure APS was to minimize the number of components in order to save weight while maintaining full fail operational/fail safe capability. Three parallel redundant regulators control LO_2 tank pressure. The valve module controlling flow through the propellant heat exchangers provides three parallel flow paths as well as series shutoff valves to control leakage. Because the Low Pressure APS can provide limit cycle operation in a simple blowdown mode without liquid vapor mixing, only double redundancy was necessary for liquid vapor mixing and engine propellant pressure regulation. The number of engines necessary to provide nominal impulse requirements allowed isolation of engines by groups after the first failure in some instances. In other areas, individual isolation for each engine is provided after first failure. A second level of isolation valves provide back-up for a double failure of an engine valve and a first level isolation valve.

To provide a basis for reliability analyses, the following criteria were established:

(1) structures, (such as lines, tanks, fittings and static seals) are assumed failure free.

(2) engines will not fail in a catastrophic mode as long as propellants are supplied at acceptable pressures and mixture ratio.

(3) normally closed shutoff valve will not fail open prior to first flight operational cycle, and leakage before first operation will be of a magnitude which will not degrade system operation.

(4) normally open shutoff valve will not fail closed prior to first flight operational cycle.

(5) liquid propellant storage tanks will not normally need venting other than that required to satisfy thermodynamic venting requirement.

(6) component external leakage can be virtually eliminated by special attention to component design details; redundancy for this failure mode was not considered in this study.

C-2. FAILURE MODE ANALYSES

Analyses were conducted to identify potential component and subassembly failure modes, their effects, and required redundancy. Detailed evaluation of each component and subassembly was made, but no attempt was made to define degrees of failures. For example, control valves were considered to have two failure modes: fail open and fail closed. Here, fail open included everything from a failure in the full open position to the lowest leakage rate affecting subsystem operation and safety.

Results of this study - the Failure Mode and Effects Analyses - are shown in Figure C-1. Each primary component, which is normally used before a failure, is identified, and potential failure modes, failure effects, failure detection requirements, and redundancy considerations are presented. No attempt was made to design an instrumentation system or to define depth of redundancy required for parameter sensing devices. Components and subassemblies are shown schematically in Figure C-2, along with a flow diagram illustrating operational mode (zero and one failure) and safe mode (two failures). Minimizing number of components and simplifying APS design were considered important goals. For example, engine isolation valves were not placed on each engine; rather, they serve specific engine groupings so that fail operational/fail safe conditions are met. Likewise, safe requirements (entry attitude control) are satisfied by simple blowdown operation, without liquid/vapor mixing, and only two regulators and liquid/vapor mixers are required. Number of components and redundancy requirements resulting from this analysis were then utilized in subsystem sizing and weight evaluations.

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: LO_2 STORAGE AND PRESSURIZATION

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
V-1 * VALVE, SHUTOFF, PNEUMATIC ACTUATION, NORMALLY CLOSED	OPENED DURING GROUND FILL OPERATION TO VENT BOIL-OFF	1 CYCLE	FAILS OPEN	NO EFFECT UNLESS V-3 ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING He TANK PRESSURE DECAY TREND.	N/C VALVE V-3 IS SERIES REDUNDANT FOR THIS FAILURE MODE AND N/O VALVE V-5 CAN BE CLOSED IN THE EVENT OF DOUBLE FAILURE OF V-1 AND V-3 IN THIS MODE.
RV-1 VALVE, RELIEF	RELIEVES LO_2 TANK OVERPRESSURE IF THE TEMPERATURE CONTROL MAL-FUNCTIONS OR REGULATOR FAILS OPEN.	72 HOURS	BURST DISK FAILS OPEN	NO EFFECT UNLESS RELIEF POPPET ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING He TANK PRESSURE DECAY TREND.	VALVE IS INTERNALLY REDUNDANT FOR THIS FAILURE MODE AND N/O VALVE V-5 CAN BE CLOSED IN THE EVENT OF DOUBLE FAILURE.
V-21 VALVE, SHUTOFF, PNEUMATIC ACTUATION, NORMALLY CLOSED	OPENED DURING GROUND FILL OPERATION FOR FILLING LO_2 TANK.	1 CYCLE	FAILS OPEN	NO EFFECT	N/C VALVES V-17 AND V-19 PROVIDE "FAIL-OP, FAIL-SAFE" REDUNDANCY.
V-23 VALVE, SHUTOFF, SOLENOID ACTUATED NORMALLY CLOSED	OPENED DURING GROUND FILL OPERA-TION FOR SERVICING He TANK.	1 CYCLE	FAILS OPEN	NO EFFECT UNLESS V-25 ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING He TANK PRESSURE DECAY TREND.	N/C VALVE V-25 IS SERIES REDUNDANT AND N/O VALVE V-15 CAN BE CLOSED IN EVENT OF DOUBLE FAILURE OF V-23 AND V-25 IN THIS MODE.
RV-3 VALVE, RELIEF	PROVIDES EMERGENCY PRESSURE RELIEF IN THE EVENT OF ACCIDENTAL OVER PRESSURE OF He TANK DURING GROUND FILL OPERATION.	1 CYCLE	BURST DISK FAILS OPEN	NO EFFECT UNLESS RELIEF POPPET ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING He TANK PRESSURE DECAY TREND	VALVE IS INTERNALLY REDUNDANT FOR THIS FAILURE MODE AND N/O VALUE V-15 CAN BE CLOSED IN EVENT OF DOUBLE FAILURE.
LO_2 TANK ASSEMBLY	PROVIDES CRYOGENIC STORAGE FOR LO_2	72 HOURS	FAILS TO MAINTAIN TEMPERATURE CONTROL	LO_2 TANK PRESSURE RISE MAY RESULT IN LOSS OF OXIDIZER BY OVERBOARD VENTING. FAILURE CAN BE DETECTED BY MONITORING LO_2 TANK TEMPERATURE AND PRESSURE.	DEPENDING UPON THE LOSS RATE, THIS FAILURE MAY REQUIRE PREMATURE REENTRY. RELIEF VALVE RV-1 WILL PROVIDE PRESSURE RELIEF. IF RV-1 FAILS CLOSED, N/C VALVES V-1 AND V-3 CAN BE OPENED TO RELIEVE PRESSURE.
R-1 REGULATOR, LO_2 TANK PRESSURE	MAINTAINS REGULATED PRESSURE IN LO_2 TANK BY ADDING HIGH PRESSURE HELIUM AS REQUIRED.	72 HOURS	FAILS CLOSED (REGULATES LOW)	LO_2 TANK PRESSURE WILL DECAY RESULTING IN DEGRADED POSITIVE EXPULSION CAPABILITY. DETECTED BY MONITORING LO_2 TANK PRESSURE.	PRESSURE REGULATORS R-3 AND R-5 PROVIDE, FAIL-OP, FAIL-SAFE" REDUNDANCY.
			FAILS OPEN (REGULATES HIGH)	EXCESSIVE LO_2 TANK PRESSURE. FAILURE DETECTED BY MONITORING LO_2 TANK PRESSURE AND TEMPERATURE.	N/O VALVES V-7 AND V-9 PROVIDE "FAIL-OP, FAIL-SAFE" REDUNDANCY FOR THIS FAILURE MODE. REDUNDANT PRESSURE SWITCHES WILL BE USED TO ISOLATE R-1 PRIOR TO CRITICAL OVERPRESSURE.

* REFERENCE SCHEMATICS ON FIGURE

FIGURE C-1

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: LH_2 STORAGE AND PRESSURIZATION

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
V-2 VALVE, SHUTOFF, PENUMATIC ACTUATION, NORMALLY CLOSED	OPENED DURING GROUND OPERATION TO VENT BOIL-OFF.	1 CYCLE	FAILS OPEN	NO EFFECT UNLESS V-4 ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING H_2 TANK PRESSURE.	N/C VALVE V-4 AND N/O VALVE V-8 PROVIDE "FAIL-OP, FAIL-SAFE" REDUNDANCY FOR THIS FAILURE MODE.
V-6 VALVE, SHUTOFF SOLENOID ACTUATED NORMALLY CLOSED	OPENED DURING GROUND OPERATION TO SUPPLY HELIUM FOR PRESSURE PAD.	1 CYCLE	FAILS OPEN	(SAME AS FOR V-2)	(SAME AS FOR V-2)
RV-2 VALVE, RELIEF	RELIEVES LH_2 TANK OVERPRESSURE IF TANK INSULATION/COOLING SYSTEM MALFUNCTIONS	1 CYCLE	BURST DISK FAILS OPEN	NO EFFECT UNLESS RELIEF POPPET ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING H_2 TANK PRESSURE.	VALVE IS INTERNALLY REDUNDANT FOR THIS FAILURE MODE AND N/O VALVE V-8 PROVIDES FAIL-SAFE REDUNDANCY IN THE EVENT OF DOUBLE FAILURE.
V-10 VALVE, SHUTOFF PNEUMATIC ACTUATION NORMALLY CLOSED	OPENED DURING GROUND FILL OPERATION TO FILL LO_2 TANK	1 CYCLE	FAILS OPEN	NO EFFECT	N/C VALVES V-12 AND V-14 PROVIDE "FAIL-OP, FAIL-SAFE" REDUNDANCY.
TANK ASSY, LH_2 STORAGE	PROVIDES CRYOGENIC STORAGE FOR LH_2	72 HOURS	FAILS TO MAINTAIN TEMPERATURE CONTROL	LH_2 TANK PRESSURE RISE MAY RESULT IN LOSS OF FUEL BY OVERBOARD VENTING. FAILURE CAN BE DETECTED BY MONITORING LH_2 TANK TEMPERATURE AND PRESSURE	DEPENDING UPON THE LOSS RATE, THIS FAILURE MAY REQUIRE PREMATURE REENTRY. RELIEF VALVE RV-2 WILL PROVIDE PRESSURE RELIEF. IF RV-2 FAILS CLOSED, N/C VALVES V-2 AND V-4 CAN BE OPENED TO RELIEVE PRESSURE.
CRYOPUMP #1, MOTOR-DRIVEN OR CRYOPUMP #2 MOTOR-DRIVEN	OPERATED TO TRANSFER LH_2 FROM THE LH_2 TANK TO THE LH_2 CONDITIONING ASSEMBLY. TWO PUMPS REQUIRED FOR NOMINAL OPERATION.		DEGRADED OUTPUT	DEGRADED LH_2 FLOW WILL DEGRADE APS OPERATION. FAILURE DETECTED BY MONITORING LH_2 FLOW RATE AT PUMP OUTLETS.	CRYOPUMP #3 PROVIDES "FAIL-OP" REDUNDANCY FOR FAILURE OF CRYOPUMP #1 OR #3. "FAIL-SAFE" REDUNDANCY IS REALIZED FROM THE FACT THAT SAFE REENTRY CAN BE ACCOMPLISHED WITH ONLY ONE PUMP OPERATIONAL.
CV-2 VALVE, CHECK OR CV-8 VALVE, CHECK	OPTIMIZES SYSTEM OPERATION BY PREVENTING BACK-FLOW THROUGH CRYOPUMP #1 WHEN NOT OPERATING.		FAILS CLOSED	NO OUTPUT FROM CRYOPUMP #1. DETECTED BY MONITORING LH_2 FLOW RATE AT PUMP OUTLET.	(SAME AS ABOVE)
CV-4 VALVE, CHECK OR CV-10 VALVE, CHECK	OPTIMIZES SYSTEM OPERATION BY PREVENTING BACK-FLOW THROUGH CRYOPUMP #2 WHEN NOT OPERATING.		FAILS CLOSED	NO OUTPUT FROM CRYOPUMP #2. DETECTED BY MONITORING LH_2 FLOW RATE AT PUMP OUTLET.	(SAME AS ABOVE)

FIGURE C-1 (Continued

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: PROPELLANT CONDITIONING (O_2 CONDITIONING SHOWN, H_2 CONDITIONING IDENTICAL EXCEPT FOR DUTY CYCLE)

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
V-29 VALVE, SHUTOFF, SOLENOID ACTUATED, NORMALLY CLOSED	CONTROLS FLOW OF LIQUID PROPELLANT TO THE HEAT EXCHANGER	87 CYCLES (O_2) 123 CYCLES (H_2)	FAILS OPEN	PROPELLANT DEPLETION RATE WILL BE EXCESSIVE. CONTINUOUS FLOW INTO BOOST TANK WILL RESULT IN EXCESSIVE VENTING. DETECTED BY MONITORING PROPELLANT FLOW INTO HEAT EXCHANGER AND BOOST TANK PRESSURE AND TEMPERATURE.	N/O VALVES V-27 AND V-31 PROVIDE "FAIL-OP, FAIL-SAFE" REDUNDANCY.
			FAILS CLOSED	NO PROPELLANT WILL FLOW TO THE HEAT EXCHANGER ALLOWING BOOST TANK PRESSURE TO DECAY. FAILURE DETECTED BY MONITORING FLOW RATE TO THE HEAT EXCHANGER.	N/C VALVES V-33 AND V-35 PROVIDE "FAIL-OP, FAIL-SAFE" REDUNDANCY FOR THIS FAILURE MODE.
R-7 REGULATOR, FLOW, MOTOR DRIVEN IRIS VALVE	CONTROLS PRESSURE OF PROPELLANT DELIVERED TO ENGINES DURING HIGH IMPULSE MANEUVERS BY MODULATING FLOW.	33 CYCLES, 800 SEC.	REGULATES LOW	INLET PRESSURE TO THE ENGINES IS LOW WITH RESULTING PERFORMANCE DEGRADATION, DETECTED BY MONITORING PROPELLANT PRESSURE UPSTREAM AND DOWNSTREAM OF THE REGULATOR.	FLOW REGULATOR R-9 PROVIDES "FAIL-OP REDUNDANCY". FAIL-SAFE REDUNDANCY IS PROVIDED BY DESIGNING THESE IRIS-TYPE VARIABLE ORIFICE VALVES TO NEVER FULLY CLOSE. APS PERFORMANCE WITH BOTH VALUES IN MINIMUM FLOW POSITION IS ADEQUATE FOR SAFE REENTRY.
			REGULATES HIGH	RESULTS IN ENGINE MIXTURE RATIO SHIFT WITH RESULTING DEGRADATION IN PROPELLANT UTILIZATION CONTROL. AFFECT ON MISSION WILL BE DEPENDENT UPON TIME OF OCCURRENCE. DETECTED BY MONITORING PROPELLANT LINE PRESSURE DOWN-STREAM OF THE VALVE.	"FAIL-OP" REDUNDANCY IS PROVIDED BY N/O VALVE V-43 IN CONJUNCTION WITH N/C VALVE V-45 AND FLOW REGULATOR R-9. "FAIL-SAFE" REDUNDANCY IS PROVIDED BY THE FACT THAT APS PERFORMANCE IS ADEQUATE FOR SAFE REENTRY WITH MAXIMUM FLOW THROUGH BOTH REGULATORS.
V-47 VALVE, SHUTOFF	CLOSED AFTER MAIN ENGINE BURN IS COMPLETED TO DIRECT PROPELLANT FLOW THROUGH THE APS DENSITY CONTROL LOOP	1 CYCLE	FAILS OPEN	ALLOWS PROPELLANT FROM BOOST TANK TO BYPASS THE DENSITY CONTROL LOOP. THIS WILL RESULT IN RAPID BOOST TANK PRESSURE DECAY DURING MAJOR BURNS WITH RESULTING ENGINE PERFORMANCE DEGRADATION. DETECTED BY MONITORING VALVE POSITION.	SHUT-OFF VALVE V-49 PROVIDES "FAIL-OP" REDUNDANCY FOR THIS FAILURE MODE. "FAIL - SAFE" REDUNDANCY IS PROVIDED BY THE FACT THAT APS PERFORMANCE IS ADEQUATE FOR SAFE REENTRY WITH BOTH VALVES FAILED FULL OPEN.
TV -1 VALVE THROTTLE MOTOR-DRIVEN	REGULATES FLOW OF LIQUID PROPELLANT TO THE LIQUID-VAPOR MIXER	33 CYCLES, 800 SEC.	FAILS CLOSED OR REGULATES LOW	PROPELLANT INLET CONDITIONS TO THE ENGINES WILL VARY DURING MAJOR BURNS CAUSING ENGINES PERFORMANCE DEGRADATION. DETECTED BY MONITORING VALVE POSITION AND PROPELLANT TEMPERATURE AND PRESSURE DOWNSTREAM OF MIXER.	THROTTLE VALVE TV-3 PROVIDES "FAIL-OP" REDUNDANCY FOR THIS FAILURE MODE. "FAIL-SAFE" REDUNDANCY IS PROVIDED SINCE APS PERFORMANCE WITHOUT CONSTANT DENSITY CONTROL IS ADEQUATE FOR SAFE REENTRY.
			FAILS OPEN OR REGULATES HIGH	PROPELLANT TEMPERATURE TO THE ENGINES LOW WITH POSSIBILITY OF SOME LIQUID BEING INJECTED OR TEMPERATURE FALLING BELOW IGNITION TEMPERATURE. DETECTED BY MONITORING PROPELLANT TEMPERATURE DOWNSTREAM OF MIXER DURING MAJOR BURNS AND BY MONITORING VALVE POSITION FOLLOW-ING THE BURN.	N/O VALVES V-37 AND V-39 PROVIDE "FAIL-OP, FAIL-SAFE" REDUNDANCY FOR THIS FAILURE MODE.

FIGURE C-1 (Continued)

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: PROPELLANT CONDITIONING (CONTINUED)

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
LIQUID VAPOR MIXER	INJECTS LIQUID PROPELLANT INTO THE GASEOUS PROPELLANT FLOW PATH TO CONTROL DENSITY OF PROPELLANT SUPPLIED TO THE ENGINES.	33 CYCLES, 800 SEC.	LIQUID INJECTOR #1 CLOGS	SAME EFFECT AS THROTTLE VALVE TV-1 FAILING CLOSED OR REGULATING LOW.	"FAIL-OP" REDUNDANCY IS PROVIDED BY LIQUID INJECTOR #3 IN CONJUNCTION WITH THROTTLE VALVE TV-3 AND VALVE V-41. "FAIL-SAFE REDUNDANCY IS PROVIDED BY THE FACT THAT APS PERFORMANCE WITHOUT CONSTANT DENSITY CONTROL IS ADEQUATE FOR SAFE REENTRY.

FIGURE C-1 (Continued)

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: PROPELLANT DISTRIBUTION AND THRUSTERS (ORBITER AND BOOSTER)

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
ENGINE ASSY. 33 ON ORBITER 20 ON BOOSTER	PROVIDES REQUIRED IMPULSE ON DEMAND	FROM 9 TO 600 CYCLES AND FROM 20 TO 500 SECONDS OF BURN TIME	FAILURE TO IGNITE OR DEGRADED PERFORMANCE	LOSS OR DEGRADATION OF THRUST WILL RESULT IN ABNORMAL COUPLING AND/OR DEGRADED ΔV. FAILURE CAN BE DETECTED BY MONITORING CHAMBER PRESSURE AND BY DETECTING ABNORMAL COUPLING MOMENTS AND ΔV.	ENGINES ARE INCORPORATED AND ARRANGED TO ALLOW FOR COMPLETE LOSS OF THRUST FROM ANY ONE ENGINE WITHOUT SIGNIFICANTLY AFFECTING APS PERFORMANCE. LOSS OF ANY TWO (MORE IN SOME CASES) MAY DEGRADE OR PREVENT COMPLETION OF THE MISSION, BUT WILL NOT PREVENT SAFE REENTRY.
VALVE, ENGINE PROPELLANT FLOW CONTROL, PNEUMATIC ACTUATION	CONTROLS ON-OFF FLOW OF PROPELLANT TO THRUST CHAMBER	SAME AS ENGINE	FAILS CLOSED	LOSS OF THRUST FROM ONE ENGINE. CAN BE DETECTED BY MONITORING VALVE POSITION AND/OR CHAMBER PRESSURE.	(SAME AS ABOVE)
			FAILS OPEN OR LEAKS	CONTINUED FLOW OF PROPELLANT TO ENGINES AFTER SHUT-DOWN WILL RESULT IN EXCESSIVE PROPELLANT CONSUMPTION. FAILURE MAY BE DETECTED BY MONITORING VALVE POSITION AND/OR PROPELLANT FLOW RATE TO EACH GROUP OF EXPECIALLY DURING IDLE PERIODS. LOW LEAKAGE RATES MAY BE DIFFICULT TO DETECT.	N/O ISOLATION VALVES PROVIDE TWO LEVELS OF ISOLATION FOR A LEAKING OR FAILED OPEN ENGINE PROPELLANT VALVE. THE FIRST LEVEL ISOLATES EITHER ONE OR A SMALL GROUP OF ENGINES WHICH WILL NOT PREVENT MISSION COMPLETION. THE SECOND LEVEL ISOLATES A LARGER GROUP OF ENGINES WHICH WILL REQUIRE MISSION TERMINATION BUT WILL NOT PREVENT A SAFE REENTRY.

FIGURE C-1 (Continued)

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: THRUSTER PNEUMATIC CONTROL - ORBITER

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
PR-1 REGULATOR, HELIUM PRESSURE OR PR-2 REGULATOR, HELIUM PRESSURE	REGULATES PRESSURE OF HELIUM SUPPLIED TO THE ENGINE VALVES FOR APPROXIMATELY ONE-HALF OF FORWARD THRUSTERS.	72 HOURS	FAILS CLOSED	TEMPORARY LOSS OF ABILITY TO CYCLE PROPELLANT VALVES OF ASSOCIATE ENGINES. DETECTED BY MONITORING DOWNSTREAM LINE PRESSURE.	VALVE PV-3 CAN BE OPENED TO ALLOW EITHER PR-1 OR PR-2 TO SUPPLY HELIUM TO ALL FORWARD ENGINE VALVES. "FAIL-SAFE REDUNDANCY IS PROVIDED BY THE FACT THAT THE APS CAN PROVIDE A SAFE REENTRY WITH ONLY THE AFT ENGINES OPERATIONAL.
			FAILS OPEN	PRESSURE TO THRUSTER VALVES WILL BE EXCESSIVE. DETECTED BY MONITORING DOWNSTREAM LINE PRESSURE. AN UNDETECTED FAILURE WILL RESULT IN LOSS OF THE FORWARD HELIUM SUPPLY THROUGH THE RELIEF VALVE.	VALVES PV-1 OR PV-2 RESPECTIVELY, PROVIDE "FAIL-OP" REDUNDANCY FOR THIS FAILURE MODE WHEN CROSSOVER VALVE PV-3 IS OPENED. "FAIL-SAFE" REDUNDANCY IS PROVIDED BY RELIEF VALVES PRV-1 AND PRV-2 AND BY THE FACT THAT THE APS CAN PROVIDE A SAFE REENTRY WITH ONLY THE AFT ENGINES OPERATIONAL.
PRV-1 VALVE, PRESSURE RELIEF OR PRV-2 VALVE, PRESSURE RELIEF	PROVIDES FOR EMERGENCY RELIEF OF LINE PRESSURE IN THE EVENT A REGULATOR FAILS OPEN AND IS NOT ISOLATED.	1 CYCLE	BURST DISK FAILS OPEN	NO EFFECT UNLESS RELIEF POPPET ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING HELIUM TANK PRESSURE DECAY RATES.	INTERNAL REDUNDANCY PROVIDES "FAIL-OP" CAPABILITY AND "FAIL-SAFE" REDUNDANCY IS PROVIDED BY THE FACT THAT THE APS CAN PROVIDE A SAFE REENTRY WITH ONLY THE AFT ENGINES OPERATIONAL.
VALVE, THRUSTER PILOT, SOLENOID ACTUATED (ANY ONE OF 33)	CONTROLS ENGINE, PROPELLANT VALVE OPERATION.	SAME AS RELATED ENGINE	FAILS CLOSED	LOSS OF ABILITY TO OPERATE ONE ENGINE. DETECTED BY MONITORING ENGINE OPERATION.	THE APS IS COMPLETELY OPERATIONAL WITH ONE
			FAILS OPEN	NO EFFECT UNLESS BACK-UP VALVE ALSO FAILS OPEN. A DOUBLE FAILURE WILL BE DETECTED AND ISOLATED BY A QUANTITY MEASURING DEVICE (FLOW) IN THE SUPPLY LINE TO EACH OF FOUR GROUPS OF ENGINES IF THE LEAKAGE RATE HIGH. LOW LEAK RATES CAN BE DETECTED BY MONITORING SUPPLY PRESSURE DECAY.	"FAIL-OP" REDUNDANCY IS PROVIDED BY A BACK-UP VALVE IN SERIES WITH EACH ENGINE PILOT VALVE. "FAIL-SAFE" REDUNDANCY IS PROVIDED BY THE THREE INDEPENDENT HELIUM SUPPLIES. SAFE REENTRY IS PROVIDED BY ANY TWO OF THE THREE SUPPLIES AND THE ASSOCIATED ENGINES. TWO FAILURES ARE REQUIRED TO LOSE ANY ONE SUPPLY.
PR-3 REGULATOR, HELIUM PRESSURE OR PR-5 REGULATOR, HELIUM PRESSURE	REGULATES PRESSURE OF HELIUM SUPPLIED TO ENGINE VALVES ONE-HALF OF THE AFT ENGINES.	72 HOURS	FAILS CLOSED	TEMPORARY LOSS OF ABILITY TO CYCLE PROPELLANT VALVES OF ASSOCIATED ENGINES. DETECTED BY MONITORING DOWNSTREAM LINE PRESSURE.	"FAIL-OP" REDUNDANCY IS PROVIDED BY A THIRD PRESSURE REGULATOR, PR-4, WHICH CAN BE USED TO REGULATE PRESSURE TO EITHER HALF, OR ALL OF THE AFT ENGINE VALVES BY USING VALVES PV-4, PV-5, PV-8 AND PV-9. "FAIL-SAFE" REDUNDANCY IS PROVIDED BY THE FACT THAT THE APS CAN PROVIDE SAFE REENTRY WITH ONLY HALF OF AFT ENGINES OPERATIONAL IF AT LEAST ONE HALF OF FORWARD ENGINES ARE OPERATIONAL.

FIGURE C-1 (Continued)

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: THRUSTER PNEUMATIC CONTROL - ORBITER (CONTINUED)

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
PR-3, PR-5 (CONTINUED)			FAILS OPEN	HELIUM PRESSURE TO ASSOCIATED ENGINE VALVES WILL BE EXCESSIVE. DETECTED BY MONITORING DOWNSTREAM LINE PRESSURE. AN UNDETECTED FAILURE WILL RESULT IN LOSS OF ONE OF THE TWO AFT HELIUM SUPPLIES THROUGH THE RELIEF VALVE.	VALVES PV-6 AND PV-7 RESPECTIVELY, PROVIDE ISOLATION CAPABILITY FOR THIS FAILURE MODE. A THIRD PRESSURE REGULATOR PR-4, IN CONJUNCTION WITH VALVES PV-4, PV-5, PV-8 & PV-9 COMPLETES "FAIL-OP" REDUNDANCY FOR THIS FAILURE MODE. "FAIL-SAFE" REDUNDANCY IS PROVIDED BY THE FACT THAT SAFE REENTRY CAN BE ACCOMPLISHED WITH ONLY ONE HALF OF AFT ENGINES FUNCTIONAL.
PRV-3 VALVE, PRESSURE RELIEF OR PRV-4 VALVE, PRESSURE RELIEF	PROVIDES FOR EMERGENCY RELIEF OF LINE PRESSURE IN THE EVENT A REGULATOR FAILS OPEN AND IS NOT ISOLATED.	1 CYCLE	BURST DESK FAILS OPEN	NO EFFECT UNLESS RELIEF POPPET ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING HELIUM TANK PRESSURE DECAY RATES.	INTERNAL REDUNDANCY PROVIDES BASIC "FAIL-OP" REDUNDANCY. FAIL-SAFE REDUNDANCY IS PROVIDED BY THE FACT THAT SAFE REENTRY CAN BE ACCOMPLISHED WITH ONLY HALF OF THE AFT ENGINES OPERATIONAL IF AT LEAST ONE HALF OF THE FORWARD ENGINES ARE OPERATIONAL.

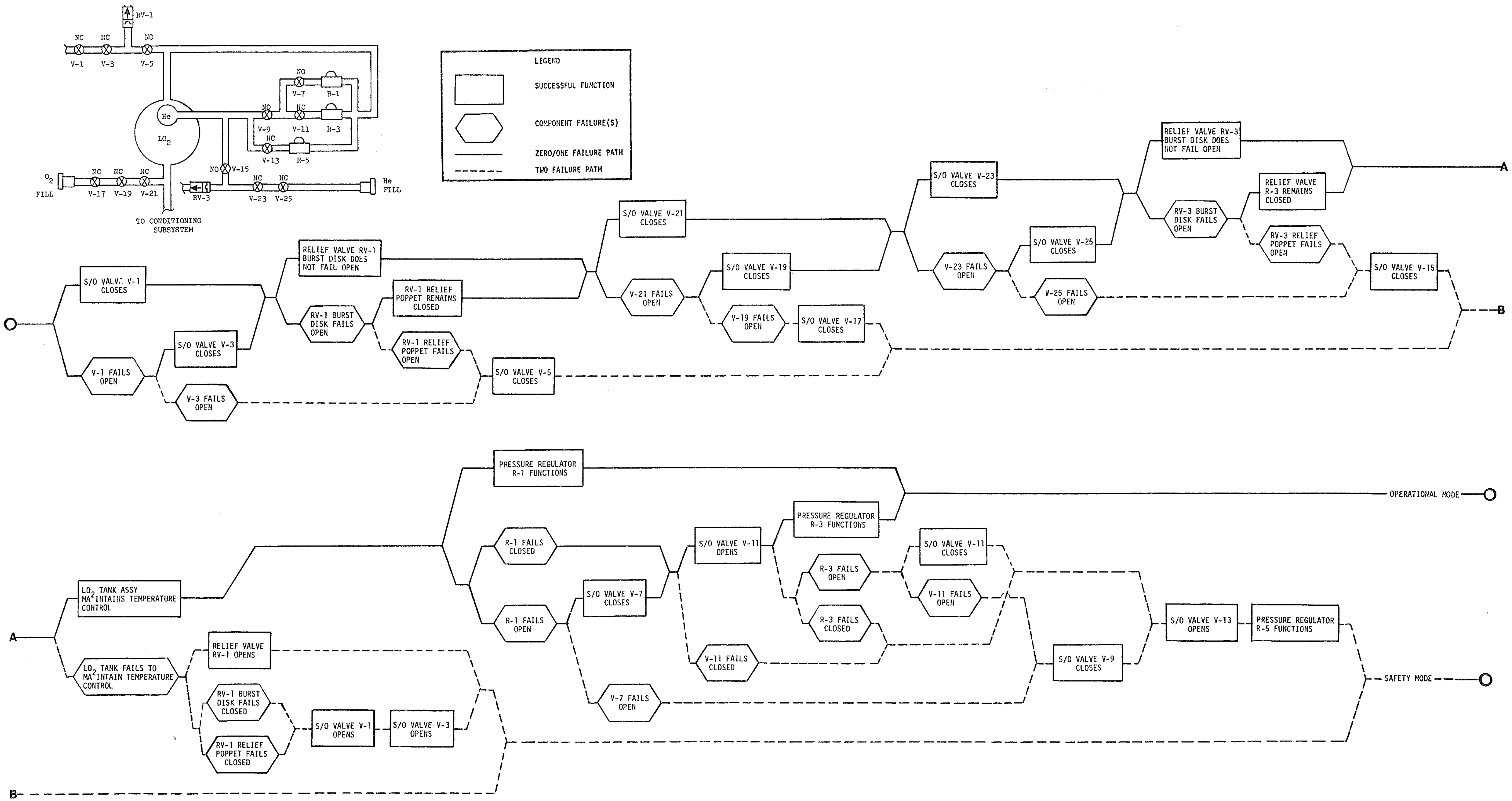
FIGURE C-1 (Continued)

FAILURE MODE AND EFFECTS ANALYSIS
PRIMARY COMPONENTS

FUNCTION: THRUSTER PNEUMATIC CONTROL - BOOSTER

COMPONENT	FUNCTION	MISSION DUTY CYCLE	FAILURE MODE	FAILURE EFFECT/DETECTION	FAIL-OP FAIL SAFE REDUNDANCY EVALUATION
PR-1 REGULATOR, HELIUM PRESSURE OR PR-2 REGULATOR, HELIUM PRESSURE	REGULATES PRESSURE OF HELIUM SUPPLIED TO PROPELLANT VALVES OF THREE GROUPS OF BOOSTER THRUSTERS	349 SECONDS	FAILS CLOSED	TEMPORARY LOSS OF ABILITY TO CYCLE PROPELLANT VALVES OF ASSOCIATED ENGINES. DETECTED BY MONITORING DOWNSTREAM LINE PRESSURE.	"FAIL-OP, FAIL-SAFE REDUNDANCY IS PROVIDED BY CROSSOVER VALVES, PV-4, PV-5, PV-6, PV-7 WHICH ALLOWS ANY ONE OF THE THREE REGULATORS TO CONTROL PRESSURE TO ALL ENGINE VALVES AND TO DRAW HELIUM FROM ALL THREE SUPPLIES.
			FAILS OPEN	HELIUM PRESSURE TO ASSOCIATED ENGINE VALVES WILL BE EXCESSIVE. DETECTED BY MONITORING DOWNSTREAM LINE PRESSURE. AN UNDETECTED FAILURE WILL RESULT IN LOSS OF ONE OF THE THREE HELIUM SUPPLIES.	VALVES PV-1, PV-2, and PV-3, REPECTIVELY PROVIDE ISOLATION CAPABILITY FOR THIS FAILURE MODE. The CROSSOVER VALVES EXPLAINED ABOVE COMPLETE THE "FAIL-OP, FAIL-SAFE" REDUNDANCY IN CONJUNCTION WITH THE FACT THAT SAFE PERFORMANCE IS POSSIBLE WITH THE LOSS OF ONE OF THE THREE SUPPLIES.
VALVE, ENGINE PILOT, SOLENOID ACTUATED	CONTROLS ENGINE PROPELLANT VALVE OPERATION	SAME AS RELATED THRUSTER	FAILS CLOSED	LOSS OF ABILITY TO OPERATE ONE ENGINE. DETECTED BY MONITORING THRUSTER OPERATION.	THE BOOSTER APS IS COMPLETELY OPERATIONAL WITH ONE ENGINE OUT.
			FAILS OPEN	NO EFFECT UNLESS BACK-UP VALVE ALSO FAILS OPEN. A DOUBLE FAILURE WILL BE DETECTED AND ISOLATED BY A QUANTITY MEASURING DEVICE (FLOW) IN THE SUPPLY LINE TO EACH OF THE THREE THRUSTER GROUPS IF THE LEAKAGE RATE IS HIGH. LOW LEAKAGE RATES CAN BE DETECTED BY MONITORING SUPPLY PRESSURE DECAY.	"FAIL-OP" REDUNDANCY IS PROVIDED BY A BACK-UP VALVE IN SERIES WITH EACH ENGINE PILOT VALVE. "FAIL-SAFE" REDUNDANCY IS PROVIDED BY THE THREE INDEPENDENT HELIUM SUPPLIES. SAFE PERFORMANCE CAN BE PROVIDED BY ANY TWO OF THE THREE SUPPLIES AND THE ASSOCIATED ENGINES.
PRV-1 VALVE PRESSURE RELIEF OR PRV-2 VALVE PRESSURE RELIEF OR PRV-3 VALVE, PRESSURE RELIEF	PROVIDES FOR EMERGENCY RELIEF OF LINE PRESSURE IN THE EVENT A REGULATOR FAILS OPEN AND IS NOT ISOLATED	1 CYCLE	BURST DISK FAILS OPEN	NO EFFECT UNLESS RELIEF POPPET ALSO FAILS OPEN. DOUBLE FAILURE CAN BE DETECTED BY MONITORING HELIUM TANK PRESSURE DECAY RATES.	INTERNAL REDUNDANCY PROVIDES "FAIL-OP" CAPABILITY AND "FAIL-SAFE" REDUNDANCY IS PROVIDED BY THE FACT THAT SAFE APS PERFORMANCE IS PROVIDED BY ANY TWO OF THE THREE HELIUM SUPPLIES AND THE ASSOCIATED ENGINES.

FIGURE C-1 (Continued)

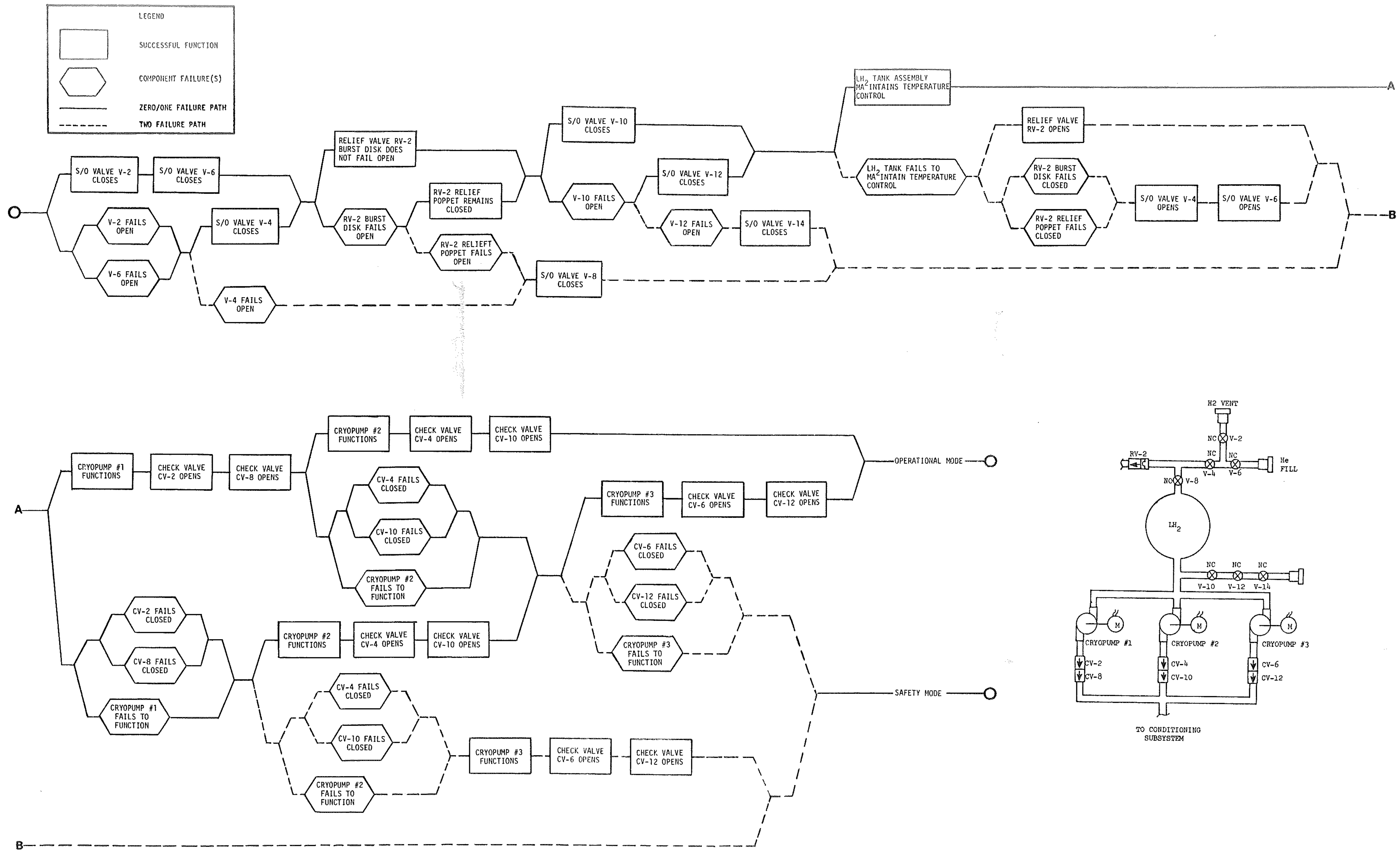


FUNCTIONAL FLOW DIAGRAM

SHUTTLE LOW PRESSURE APS-LO_2 STORAGE AND PRESSURIZATION

ORBITER

FIGURE C-2

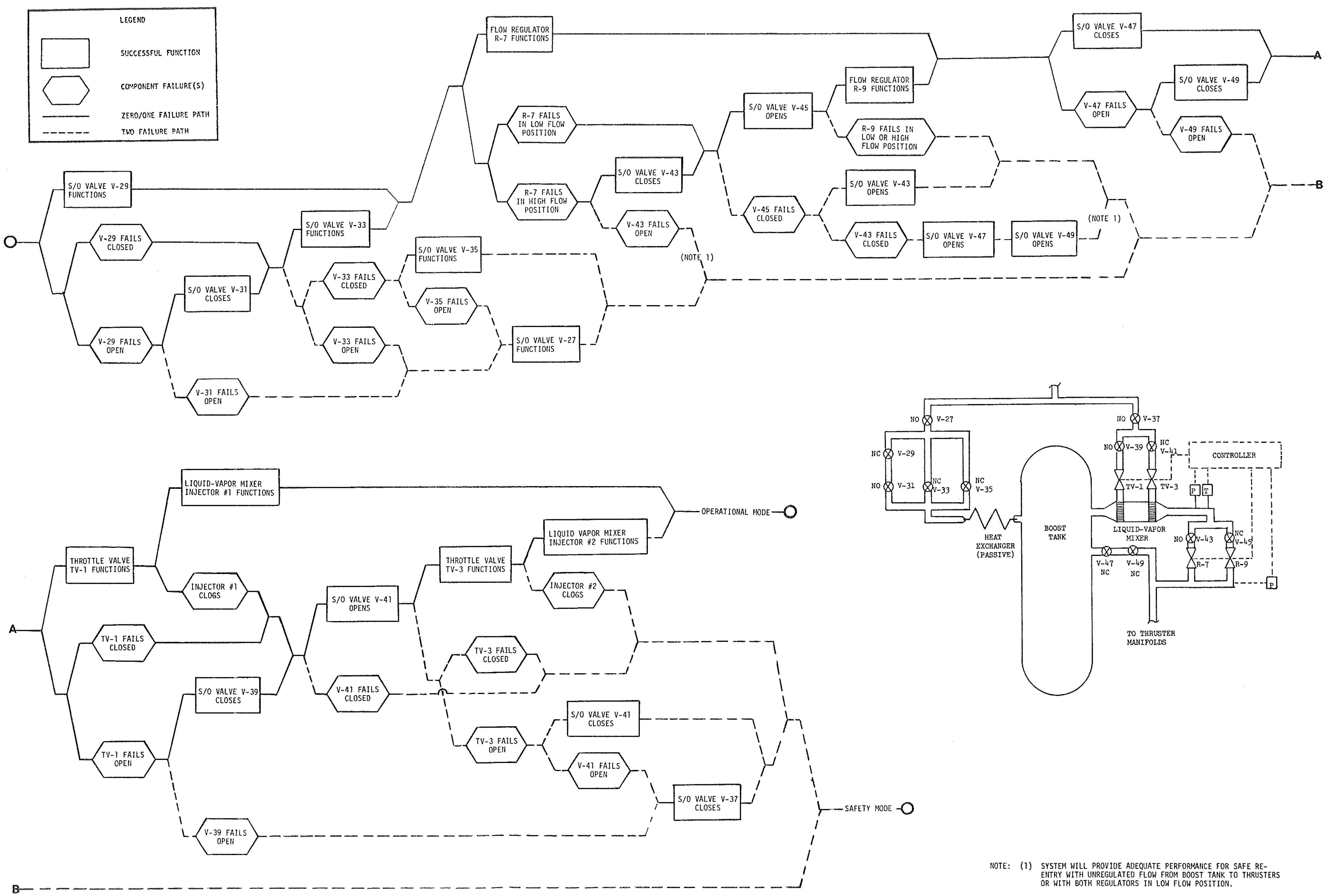


FUNCTIONAL FLOW DIAGRAM

SHUTTLE LOW PRESSURE APS-LH_2 STORAGE AND PRESSURIZATION

ORBITER

FIGURE C-2 (Continued)



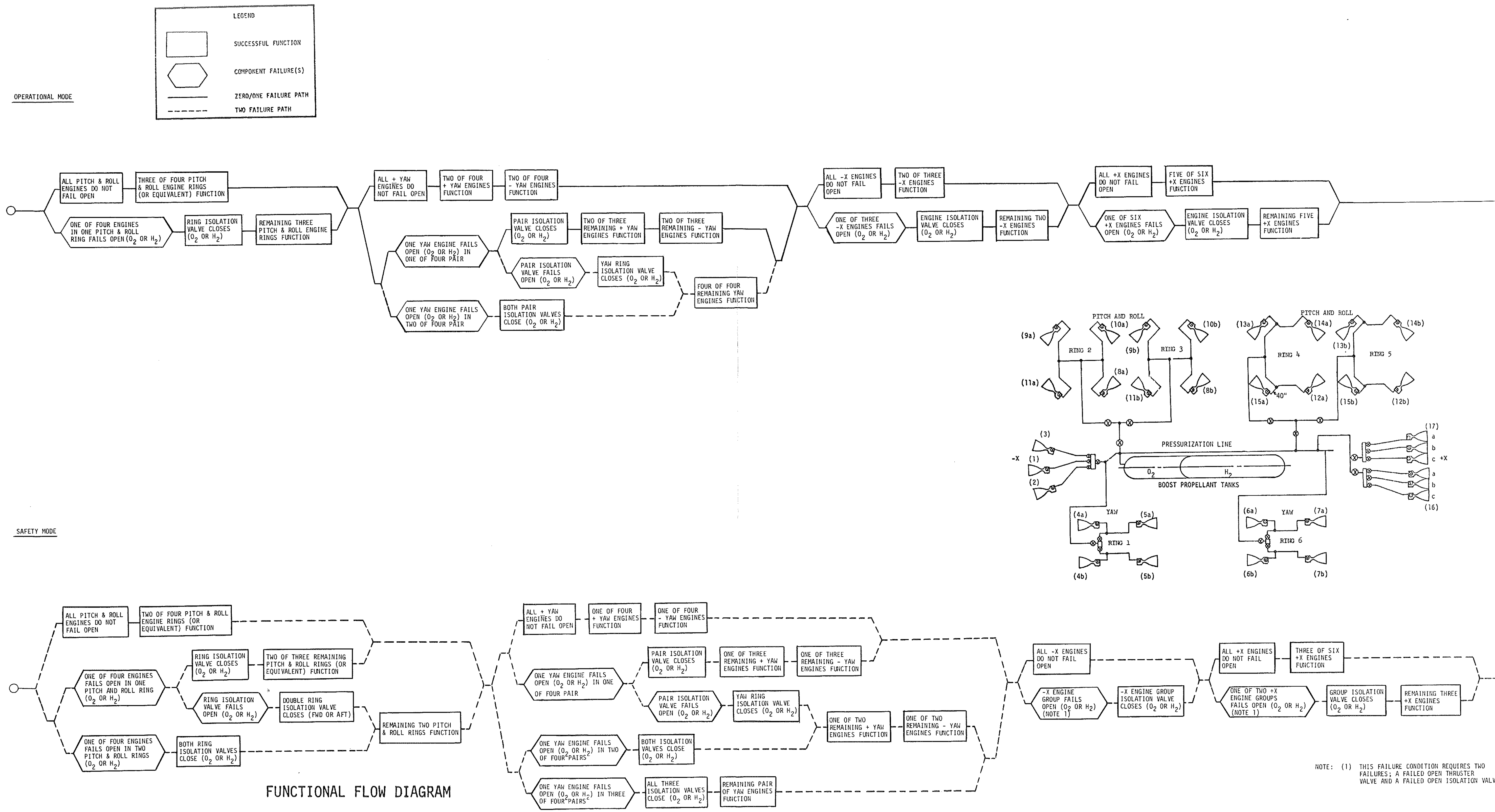
NOTE: (1) SYSTEM WILL PROVIDE ADEQUATE PERFORMANCE FOR SAFE RE-ENTRY WITH UNREGULATED FLOW FROM BOOST TANK TO THRUSTERS OR WITH BOTH REGULATORS IN LOW FLOW POSITION.

FUNCTIONAL FLOW DIAGRAM

SHUTTLE LOW PRESSURE APS-PROPELLANT CONDITIONING

ORBITER

FIGURE C-2 (Continued)



FUNCTIONAL FLOW DIAGRAM

SHUTTLE LOW PRESSURE APS-PROPELLANT DISTRIBUTION AND ENGINES

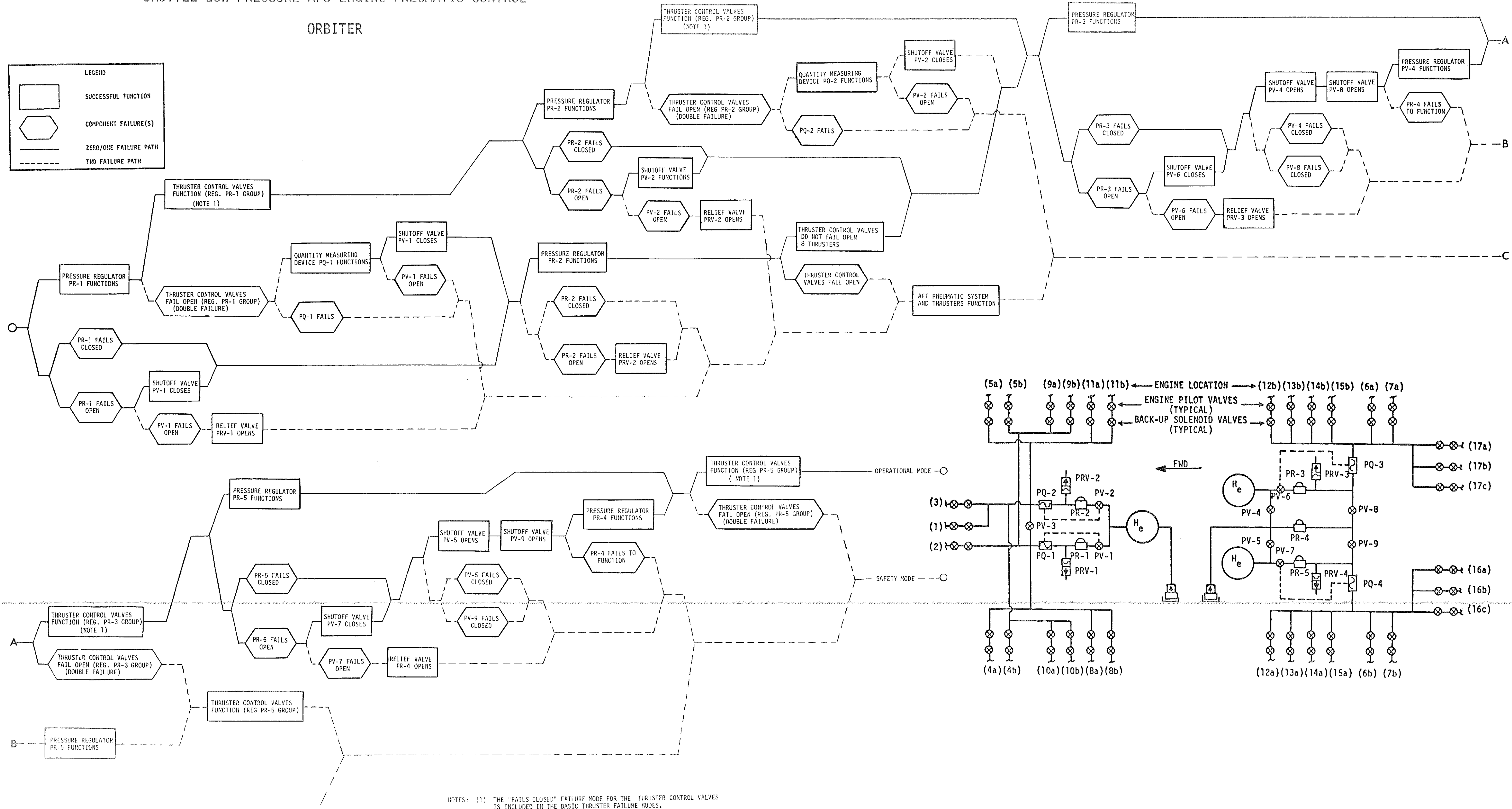
ORBITER

FIGURE C-2 (Continued)

FUNCTIONAL FLOW DIAGRAM

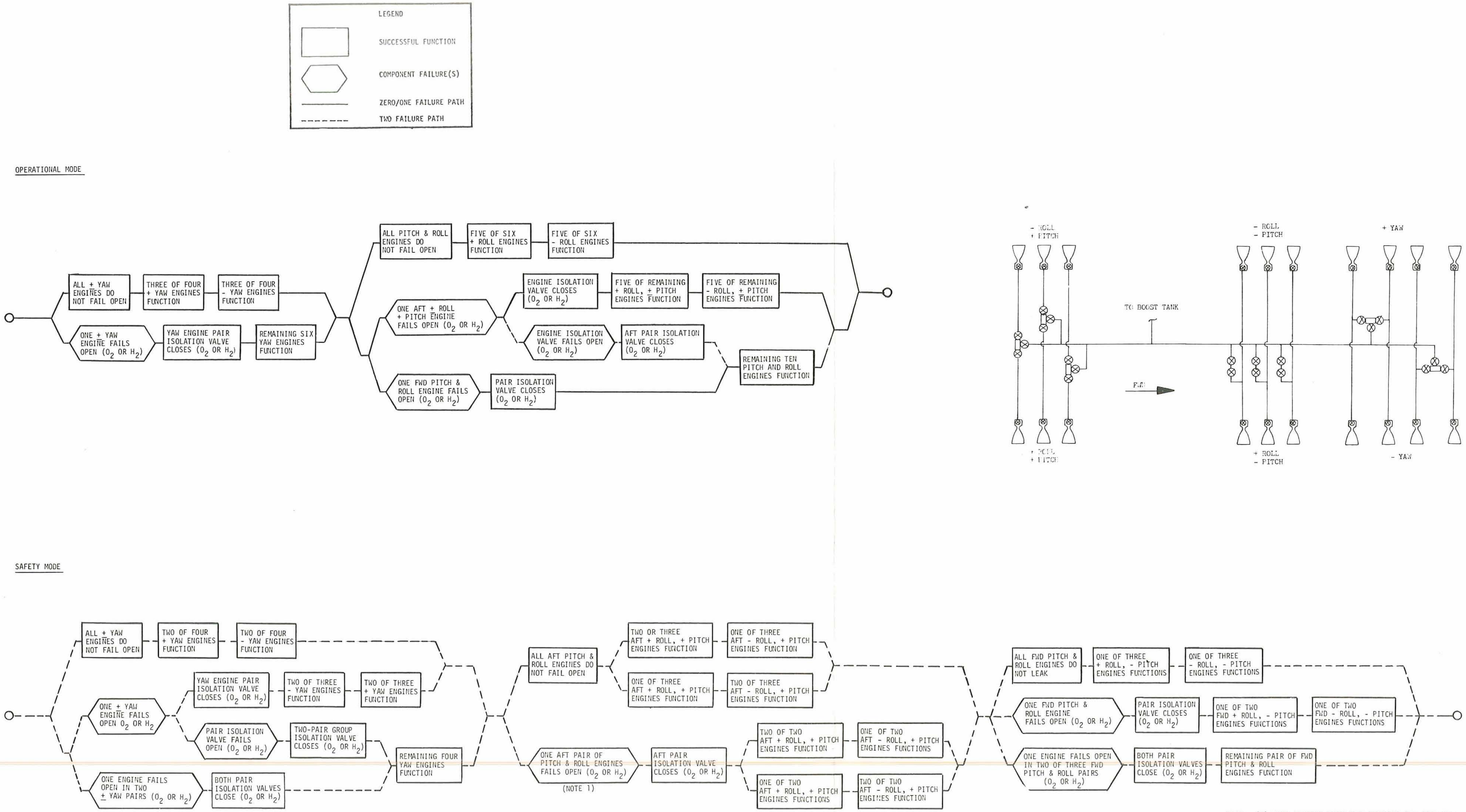
SHUTTLE LOW PRESSURE APS-ENGINE PNEUMATIC CONTROL

ORBITER



NOTES: (1) THE "FAILS CLOSED" FAILURE MODE FOR THE THRUSTER CONTROL VALVES IS INCLUDED IN THE BASIC THRUSTER FAILURE MODES.

FIGURE C–2 (Continued)



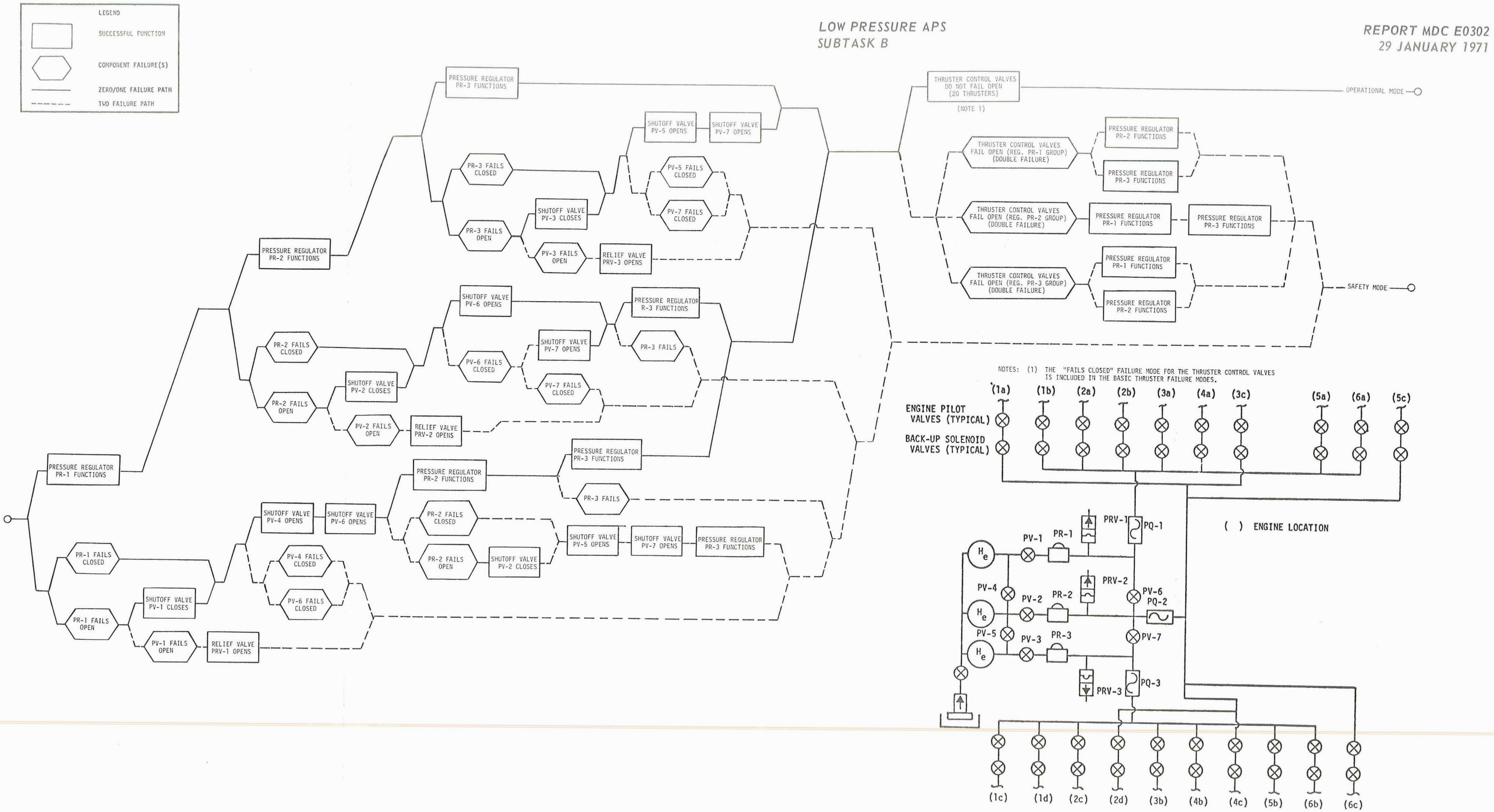
NOTE: (1) THIS FAILURE CONDITION REQUIRES TWO FAILURES; A FAILED OPEN THRUSTER VALVE AND A FAILED OPEN ISOLATION VALVE.

FUNCTIONAL FLOW DIAGRAM

SHUTTLE LOW PRESSURE APS-PROPELLANT DISTRIBUTION AND ENGINES

BOOSTER

FIGURE C-2 (Continued)



FUNCTIONAL FLOW DIAGRAM

SHUTTLE LOW PRESSURE APS-ENGINE PNEUMATIC CONTROL

BOOSTER

FIGURE C-2 (Continued)

C-3. RELIABILITY ESTIMATES

A measure of potential APS "numerical" reliability was derived. Reliability estimates were, in general, determined by applying component failure rates and duty cycles listed in Figure C-3 to subsystem functional flow diagrams (presented above). In some instances, approximation techniques were used to compute the estimate for redundant and complex component groups. The most common approximation used was a calculation of "Fail Safe" probability as:

$$P(FS) = 1 - \sum P \text{ (of each combination of failures occuring which create a potential hazard.)}$$

In view of the preliminary nature of component and subsystem design, accuracy of these approximations is well within the failure rate accuracy used in the analyses. Failure rates, as listed in Figure C-3, were selected after a review of available failure rate data. Percentage of failure rate assigned to each component failure mode is a judgment figure based on review of failure data (where available). Duty cycles shown for each component cover a 72 hour mission time line. Environmental and operational factors were not applied to these duty cycles because components are generally designed for much more severe environments that those encountered in actual usage.

The results of these estimates, presented in Figure C-4, indicate that the low pressure APS baseline designs described in this report offer high potential reliability.

SHUTTLE LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM
COMPONENT DUTY CYCLES FAILURE MODES AND FAILURE RATES

ORBITER AND BOOSTER

COMPONENT TYPE	IDENTIFICATION NUMBERS (SCHEMATIC)	MISSION DUTY CYCLE	FAILURE MODE	FAILURE RATE X 10^6	SOURCE OF BASIC FAILURE RATE
LIQUID-VAPOR MIXER		33 CYCLES 800 SECONDS	INJECTOR CLOGS	NEGLIGIBLE	---
PUMP, CRYOGENIC, MOTOR DRIVEN	CRYOPUMP #1 CRYOPUMP #2 CRYOPUMP #3	123 CYCLES 900 SECONDS	INOPERATIVE	5.0/CYCLE + 50/HOUR	CYCLIC - ESTIMATE HOURLY - PESCO PRODUCTS DATA
REGULATOR, PRESSURE, HELIUM	R-1, R-3, R-5, PR-1, PR-2, PR-3, PR-4, PR-5	72 HOURS (ORBITER) 0.1 HOURS (BOOSTER)	FAILS OPEN FAILS CLOSED	30/HOUR 11/HOUR	MDAC GEMINI EXPERIENCE
REGULATOR, PRESSURE, MOTOR DRIVEN IRIS VALVE	R-7, R-9	33 CYCLES 800 SECONDS	FAILS IN LOW FLOW POSITION FAILS IN HIGH FLOW POSITION	50/HOUR 25/CYCLE 50/HOUR 25/CYCLE	BASIC FAILURE RATE ASSUMED THE SAME AS FOR MOTOR DRIVEN THROTTLING VALVE
VALVE, CHECK LH_2	CV-2, CV-4, CV-6, CV-8, CV-10, CV-12	123 CYCLES EACH	FAILS OPEN FAILS CLOSED	2.2/CYCLE 0.1/CYCLE	AVCO RELIABILITY ENGINEERING DATA SERIES, FAILURE RATES, APRIL 1962
VALVE, S/O, PNEUMATIC ACTUATION, NORMALLY CLOSED, FILL & DRAIN	V-1, V-2, V-3, V-4, V-10, V-12, V-14, V-19, V-21	1 CYCLE EACH	FAILS OPEN FAILS CLOSED	510/CYCLE 100/CYCLE	FARADA - SATURN
VALVE, S/O, MOTOR DRIVEN, ISOLATION	V-5, V-15, V-43, V-45, V-47, V-49, PLUS ALL ENGINE ISOLATION VALVES	1 CYCLE EACH AS REQUIRED	FAILS OPEN FAILS CLOSED	16/CYCLE 3.3/CYCLE	NASA DATA-SATURN (SHUTOFF VALVE)
VALVE, LIQUID THROTTLING MOTOR DRIVEN	TV-1, TV-3	33 CYCLES 800 SEC	FAILS OPEN FAILS CLOSED	80/HOUR + 40 CYCLE 20/HOUR + 10 CYCLE	HOURLY - LTV ELECTROSYSTEMS DATA CYCLIC - STANFORD RESEARCH INSTITUTE CONTRACT NAS 7-751 (ELECTROMECHANICAL ACTUATOR)
VALVE, SOLENOID ACTUATED, HELIUM CONTROL	V-6, V-7, V-9, V-11, V-13, V-15, V-25, PV-1, PV-2, PV-3, PV-4, PV-5, PV-6, PV-7, PV-8, PV-9	1 CYCLE EACH AS REQUIRED	FAILS OPEN FAILS CLOSED	8.0/CYCLE 0.77/CYCLE	THIOKOL CHEMICAL CORP. (SURVEYOR)
	ENGINE PILOT VALVES AND BACK-UP VALVES	SAME AS THRUSTERS			
VALVE, SOLENOID ACTUATED, LIQUID SHUTOFF	V-31, V-37, V-39, V-41	1 CYCLE EACH AS REQUIRED	FAILS OPEN FAILS CLOSED	5.8/CYCLE 0.56/CYCLE	NASA DATA - SATURN
	V-29, V-33	123 CYCLES (H_2) 87 CYCLES (O_2)			
	V-35, V-27	18 CYCLES - REENTRY			
VALVE, RELIEF, PRESSURE	RV-1, RV-2, RV-3, PRV-1, PRV-2, PRV-3, PRV-4	1 CYCLE EACH AS REQUIRED	FAILS OPEN FAILS CLOSED	5.3/CYCLE 1.5/CYCLE	NASA DATA - SATURN
(BURST DISK)	RV-1, RV-2, RV-3, PRV-1, PRV-2, PRV-3, PRV-4		FAILS OPEN FAILS CLOSED	60/UNIT 40/UNIT	STANFORD RESEARCH INSTITUTE CONTRACT NAS 7-751
TANK ASSEMBLY, LIQUID PROPELLANT	LO_2 TANK LH_2 TANK	72 HOURS	FAILS TO MAINTAIN TEMPERATURE CONTROL	NEGLIGIBLE	---
ENGINE ASSEMBLY FILM COOLED (INCLUDING VALVES)	ORBITER + X - X PITCH & ROLL YAW	24 CYCLES 9 CYCLES 650 CYCLES 650 CYCLES EACH ENGINE	FAILS TO FUNCTION FAILS OPEN OR LEAKS	5.0/CYCLE 7.5/CYCLE	MDAC ESTIMATE-AFTER DISCUSSION WITH ENGINE MANUFACTURERS (AEROJET, BELL, MARQUARDT, AND ROCKETDYNE)
	BOOSTER PITCH & ROLL YAW	100 CYCLES 40 CYCLES EACH ENGINE			
QUANTITY MEASURING DEVICE	PQ-1, PQ-2, PQ-3, PQ-4	72 HOURS (ORBITER) 0.1 HOURS (BOOSTER)	FAILS TO FUNCTION	2.5/HOUR	AVCO RELIABILITY ENGINEERING DATA SERIES, FAILURE RATES, APRIL 1962

FIGURE C-3

FUNCTIONAL GROUP	ORBITER OPERATIONAL	ORBITER FAIL SAFE	BOOSTER OPERATIONAL	BOOSTER FAIL SAFE
LO_2 STORAGE AND PRESSURIZATION	0.999990	0.99999997	–	–
LH_2 STORAGE AND PRESSURIZATION	0.999997	0.99999999	–	–
PROPELLANT CONDITIONING – O_2	0.999994	0.99999999	–	–
PROPELLANT CONDITIONING – H_2	0.999993	0.99999999	–	–
PROPELLANT DISTRIBUTION AND ENGINES	0.999743	0.99997069	0.999923	0.99999998
ENGINE PNEUMATIC CONTROL	0.999749	0.99999996	0.999991	0.99999999
SUBSYSTEM	0.997196	0.99997059	0.999914	0.99999997

ASSUMPTIONS:
(1) COMPONENT EXTERNAL LEAKAGE CAN BE CONTROLLED BY PROPER SEAL DESIGN
(2) SENSING AND SWITCHING RELIABILITY IS EQUAL TO 1.0.
(3) STRUCTURAL RELIABILITY IS EQUAL TO 1.0.
(4) MAIN PROPULSION SUBSYSTEM COMPONENTS USED BY THE AUXILIARY PROPULSION SYSTEM WILL NOT DEGRADE APS OPERATION OR SAFETY.
(5) THE NON–OPERATING FAILURE RATE FOR APS COMPONENTS WILL NOT BE SIGNIFICANT

APS RELIABILITY

FIGURE C–4

APPENDIX D TANKAGE

D-1. PROPELLANT STORAGE, ACQUISITION, AND PRESSURIZATION

During Subtask A, analyses and trade studies were conducted to identify preliminary propellant storage assembly selections. It was concluded that separate APS tankage was more attractive than integrated APS/OMS tankage. Also, a simple, regulated pressure, helium pressurization assembly was judged better for oxygen tanks and an autogenous pressurization assembly, using a compressor, was more attractive for hydrogen tankage. Component models used to conduct Subtask A analyses were not sophisticated, and (especially in the case of the positioning and vent assemblies) design details were not important since these would have little effect on subsystem weight or technology effecting selection of preferred approaches. Therefore, during Subtask B, it was necessary to conduct the detailed analyses required to define more accurately design and performance of propellant tankage assemblies and to confirm and/or update the propellant integration approach selected in Subtask A.

For these analyses, baseline tank sizes were established, based on Subtask A requirements, alternate design approaches for different tankage assemblies were investigated, preferred approachs were selected, and design and performance characteristics were established. This appendix defines baseline requirements used for Subtask B analyses and summarizes trade studies effecting definition of assembly design for propellant acquisition, insulation, cooling, and pressurization. In addition, this appendix defines results of an updated integration study and provides a summary of selected low pressure APS tankage design.

D-1.1 Baseline Requirements - Significant APS tankage requirements affecting preliminary design are shown in Figure D-1. These requirements were based principally on Subtask A results, defined in Reference (a). Spherical tanks with diameters of 10.6 ft for hydrogen and 5.04 ft for oxygen were required for APS propellant storage. OMS tank requirements are also shown. These were based on use of an RL10A-3-3A engine in the OMS subsystem and OMS maneuver velocity allocation of 1150 ft/sec. Analysis of the various tankage assemblies based on these requirements, review of the assembly options, definition of concepts selected, and definition of their operating characteristics is provided below.

PROPELLANT WEIGHT - LB	HYDROGEN	OXYGEN
APS	2499	4496
OMS	3775	18,943
TOTAL APS PLUS OMS	6274	23439
APS TANKAGE CHARACTERISTICS		
NO TANKS	1	1
TANK SHAPE	SPHERICAL	SPHERICAL
VOLUME - FT^3	634	67
DIAMETER - FT	10.6	5.04

APS TANKAGE REQUIREMENTS

FIGURE D-1

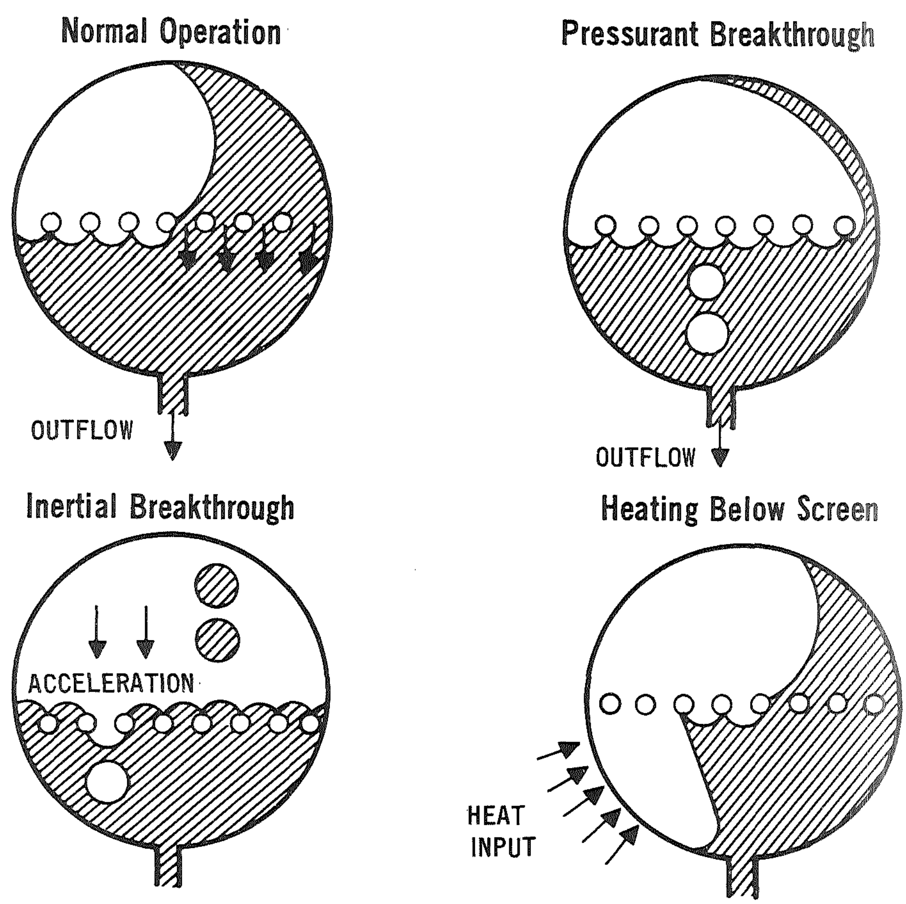
D-2. PROPELLANT ACQUISITION ASSEMBLY

A propellant positioning device is required in the APS tankage to ensure liquid outflow during low g orbital phases of the mission. During these phases, vehicle accelerations tend to randomly orient bulk propellant mass within the tank with the potential of uncovering the tank outlet and causing a loss of pressurant gas and interruption of liquid flow. Some device is required to either totally constrain the liquid at the tank outlet or to provide a path of communication from liquid mass to tank outlet. Total constraint by positive expulsion was impractical because of tank size, number of cycles required for reuse, and the cryogenic properties of the propellant. The only reasonable approach available was to use a surface tension screen device to provide a flow path to the tank outlet.

Surface tension devices had been successfully used on several vehicles, including Agena, Apollo, Transtage, and drone and target aircraft. Also, these devices had been subject to extensive laboratory testing. They are passive in nature, and have no moving parts, resulting in high reliability and multicycle reuse capability. Sufficient design information was available to establish, with high confidence, that a surface tension assembly could be designed for orbiter application.

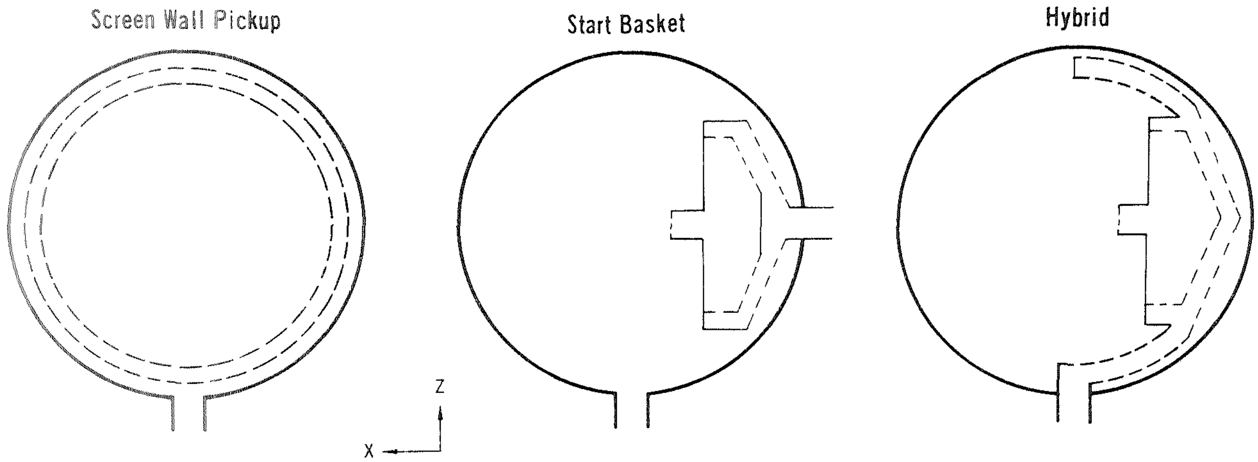
Basic physical processes associated with surface tension concepts are shown in Figure D-2. Under normal operation, the screen is completely wetted on one side. If gas contacts the screen on the other side, surface tension forces prevent movement of the vapor through the screen. When liquid is in contact on the other side, liquid is free to travel from one side to the other as the surface tension effect is not present. Thus, successful acquisition of the liquids will be achieved until: (a) there is no liquid/liquid interface; (b) acceleration forces of sufficient magnitude to exceed surface tension pressure capability are present; or (c) heating below the screen causes vaporization on the liquid side, hence a gas/gas interface across the screen. In this last case, the surface tension screen would preferentially flow pressurant gas.

D-2.1 Acquisition Concepts and Selection - Three basic options for acquisition assembly designs are available. These are illustrated schematically in Figure D-3. The first acquisition device consists of a screen configured to locate the screen surface close to the tank wall (i.e., a wall-oriented screen). With this approach, liquid withdrawal is possible as long as liquid is in contact with the tank wall. This approach is, therefore, mission-independent, since the specified liquids are wetting and will always assume a wall-contact orientation.



SURFACE TENSION DEVICE OPERATION

FIGURE D-2



Screen Wall Pickup

1. MUST BE LAUNCHED FULL
2. MISSION-INDEPENDENT. NO SETTLING REQUIRED
3. SINGLE TANK OUTLET
4. LARGE SIZE REQUIRES FINE SCREEN MESH
5. LOW-G TRANSFER CAPABILITY

Start Basket

1. PARTIAL FULL LAUNCH POSSIBLE
2. TWO TANK OUTLETS REQUIRED
3. MISSION-DEPENDENT. SETTLING REQUIRED
4. LIQUID "FALLOUT" DURING OUTFLOW IN LATERAL ACCELERATION

Hybrid

1. PARTIAL FULL LAUNCH POSSIBLE
2. SINGLE TANK OUT
3. BASKET SIZED TO PROVIDE LIQUID WHEN LIQUID IS IN -X END OF TANK
4. SOME SETTLING STILL REQUIRED
5. NO "FALLOUT" DURING LATERAL ACCELERATION

ACQUISITION DEVICE CANDIDATES

FIGURE D-3

The second device is a start basket. This approach is mission-dependent, because it must be refilled by translation accelerations periodically during the mission. It also requires two tank outlets since the large OMS thrust level will settle propellant in the -X end of the tank (refilling the basket) while entry accelerations will settle the propellant in the -Z portion of the tank.

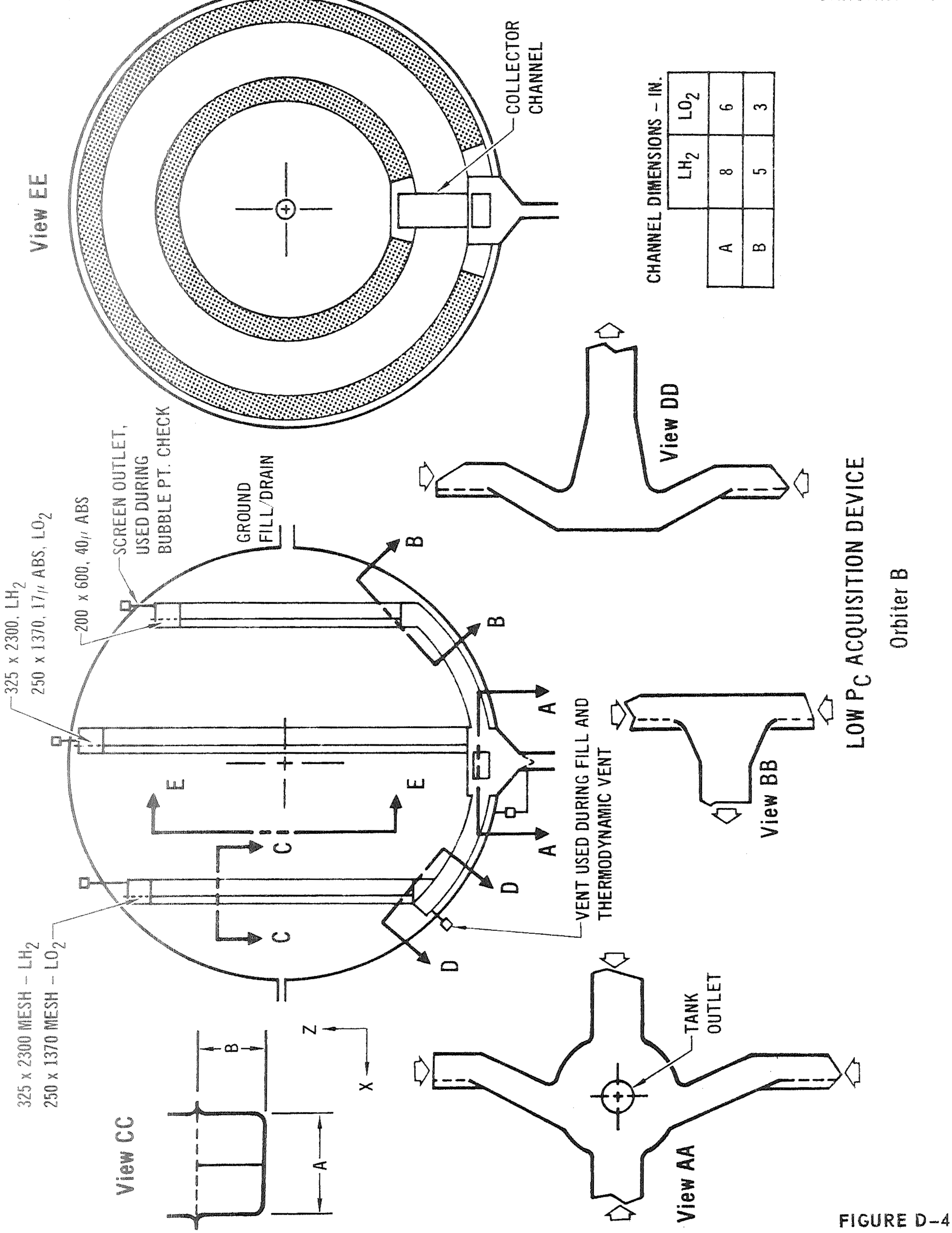
The third acquisition candidate is a hybrid assembly, combining the start basket with a wall-oriented approach. Only a single tank outlet is needed, but the device is still mission-dependent since the basket must be refilled periodically. The wall oriented device was selected because it is mission-independent, does not require two outlets, and needs no refill.

Several alternates could be conceived for a wall oriented positioning approach. The most conventional of these is a continuous tank liner made of screen. This approach has received the most attention in exploratory development testing, but was considered to be impractical for a large tank because of fabrication difficulty and boost ullage considerations (Reference (b)). An alternate, and more practical, configuration is a screen channel device, illustrated in Figures D-4 and D-5. Several aluminum channels or annular trays are located within the propellant tank in close proximity to the wall. These assure contact between liquid and positioning device under any random orientation conditions. The device is insensitive to boost accelerations, since trays are submerged during boost, and it is the most practical design approach from fabrication and screen size standpoints. In addition, the design readily provides capability of checking screen bubble point pressure after complete tank assembly.

D-2.2 Design - The propellant acquisition device consists of three screen channels, or trays, around the circumference of the propellant tank and a single, enclosed collector channel which connects each annular tray to the sump (Reference Figures D-4 and D-5). The annular trays are normal to the vehicle longitudinal (or +X) axis and the sump is located in the tank bottom (or the -Z) extremity of the tank. The tank outlet has a cone-shaped vapor/liquid interface screen located within the feed line below the sump. The feed line is connected to the propellant tank by a small vapor relief line, allowing any vapor developed in the line to pass back into the tank vapor region. Solid portions of the trays are formed from 0.02 in. aluminum sheet. The center wall in the channel is present to increase box cross section rigidity. It is anticipated that channels would be fabricated in quarter sections, inspected, and then joined together inside the tank. Each quarter section has two points of attachment to the tank wall. At these points, then, low conductivity fiberglass rods would offer channel support.

D-2.3 Operation - It is a requirement that screen surfaces must contact the bulk liquid throughout the mission and that channels remain completely full of liquid at all times. During outflow, the acquisition device will selectively pass liquid to the feed system if it is in contact with a liquid mass. The wall-oriented nature of the screen device ensures that contact will be made. Screen mesh and flow passage dimensions are selected so that pressure drop across screen vapor/liquid interface never exceeds screen bubble point prior to reentry. Two different mesh sizes have been selected for the screen channels. A relatively coarse mesh can be used in the screen channel near the aft end of the vehicle. The other two channels require a finer mesh, to withstand hydrostatic head during periods of high maneuver accelerations and low propellant loading.

Placement of the screen channel rings within the propellant tank was based on propellant quantities prior to entry. Approximately 13 percent of propellant will be in the APS tank prior to entry and will be settled in the -X end of the tank by the OMS thrust during the deorbit burn. The bottom screen channel is placed just below the bulk liquid surface at this propellant loading. Subsequent to deorbit burn, propellant will be reoriented to the tank sump by entry drag forces,



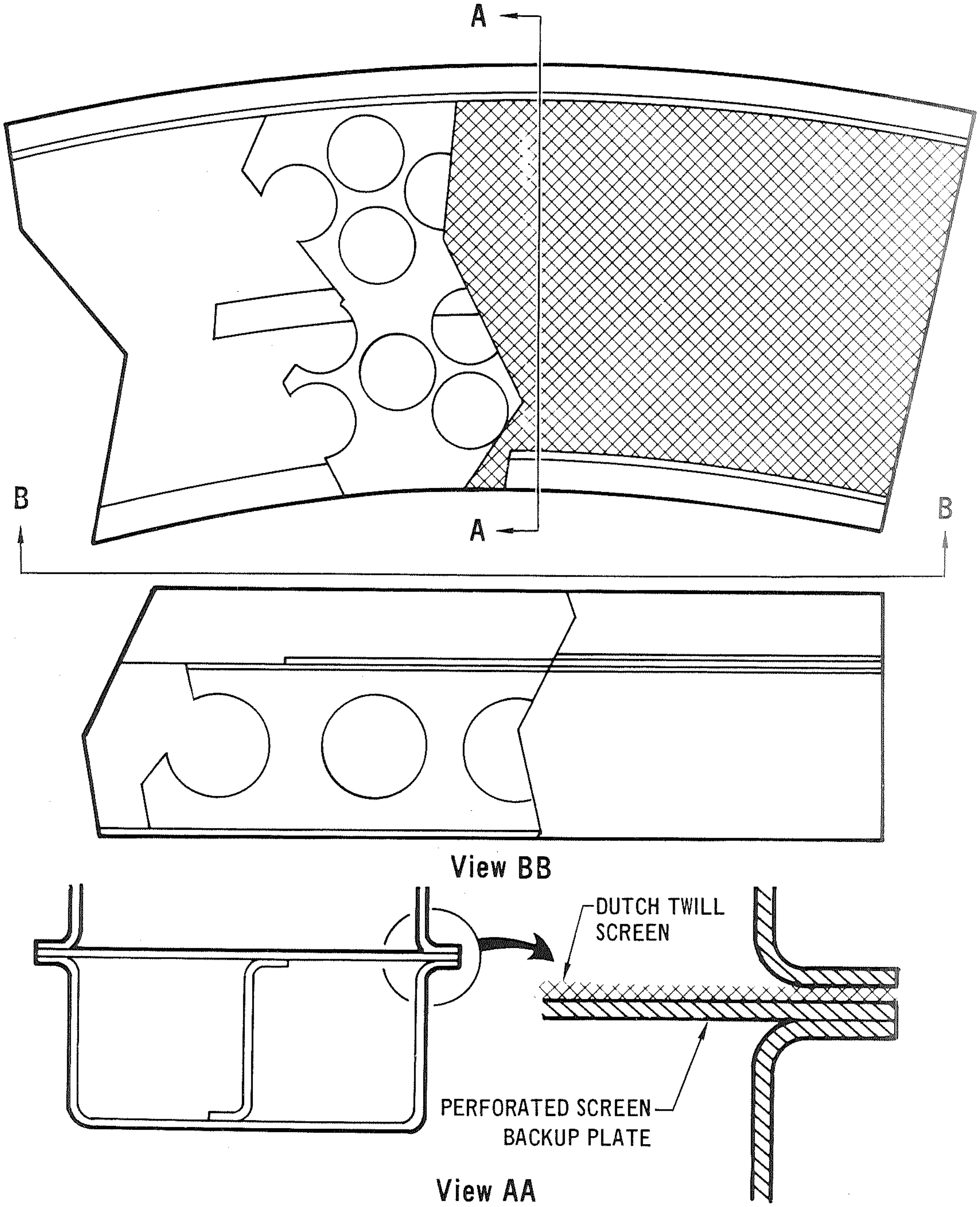
CHANNEL DIMENSIONS – IN.

	LH_2	LO_2
A	8	6
B	5	3

LOW P_C ACQUISITION DEVICE

Orbiter B

FIGURE D-4



PROPELLANT ACQUISITION SCREEN CHANNEL DESIGN

FIGURE D-5

and therefore, will remain in contact with one or both of the remaining screen channels. High +Z acceleration levels during entry will exceed the stability limit of the acquisition device. When this occurs, pressurant will enter the screen channels and levels within channels and collector will quickly drop until they approximately match the liquid level in the tank bottom.

D-3. INSULATION

Considerable research has been, and is being, devoted to development of high performance insulation (HPI) concepts. In general, HPI concepts utilize sheets of highly reflective metalized plastic film (such as aluminized mylar) made into blankets. Separation of sheets in the blankets is provided by embossing, or flocking, the basic film material, or by using a separator sheet such as netting or glass fabrics. Many design variations are possible, and separate technology studies are currently underway to establish optimum shuttle HPI designs. For purposes of this study, however, differences in alternate HPI approaches would have little effect on overall storage weight. For this reason, a typical HPI concept was selected and used to optimize amount of insulation and to define reasonable propellant storage and vent losses.

D-3.1 Insulation Selection/Optimization - MDAC (under an insulation technology study for NASA, Contract No. NAS 8-21400) has investigated various HPI systems. Based on this effort, typical insulation characteristics were selected for APS tankage analysis. The selected scheme is made up of double aluminized mylar (DAM) reflectors of 0.15 mil thickness with a dacron net separator. A density of 5 lb/ft^3 based on a sheet density of 90 sheets per inch is representative of this insulation. This insulation has been experimentally evaluated through calorimeter test of blanket samples. Figure D-6 shows basic insulation characteristics and effective conductivity degradation caused by perforations, joints, and attachments.

The data of Figure D-6, together with baseline propellant requirements, were used to optimize HPI thickness for the shuttle mission. Insulation thickness was based on providing a maximum storage efficiency, (i.e., ratio of usable propellant to total storage assembly weight). All heat was assumed to go into propellant vaporization and boiloff. This assumption is in accordance with the thermal vent/shroud system which converts the heat input into hydrogen vaporization. Resulting optimum number of insulation layers and total weight of propellant vented, assuming no heat shorts, are shown in Figure D-7 (as a function of orbit time). Optimum insulation for the hydrogen tank had 62 sheets, and was 0.68 in thick, weighing 85 lb. This resulted in a hydrogen vent rate of 0.37 lb/hr.

D-3.2 Reusability Characteristics - One of the major concerns in design of the insulation is the reusability requirement: high confidence can be placed in the prediction of basic HPI heat transfer characteristics and thermal performance.

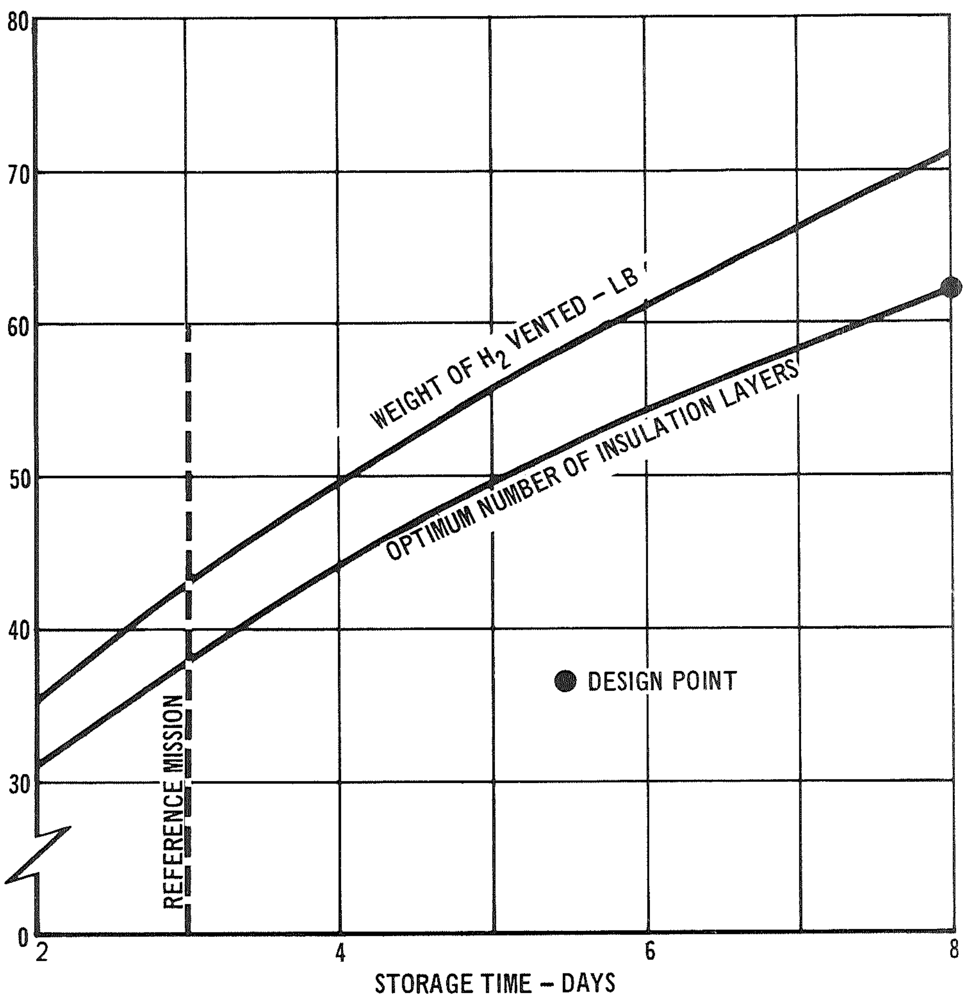
DOUBLY ALUMINIZED MYLAR (DAM) – DACRON NET SEPARATOR

15 IN LH_2 CALORIMETER TESTING – $T_H = 530^oR$

	EFFECTIVE CONDUCTIVITY – $\frac{BTU-FT}{FT^2-HR-^oR}$
BASIC INSULATION BLANKET	1.37×10^{-5}
WITH 2.3% PERFORATIONS	1.69×10^{-5}
WITH PERFORATION AND JOINT	2.03×10^{-5}
WITH PERFORATION JOINT AND ATTACHMENT	2.217×10^{-5}

MULTI-LAYER INSULATION MEASURED PERFORMANCE CHARACTERISTICS

FIGURE D-6



OPTIMUM LH_2 TANK INSULATION
(Assuming No Tank Pressure Rise)

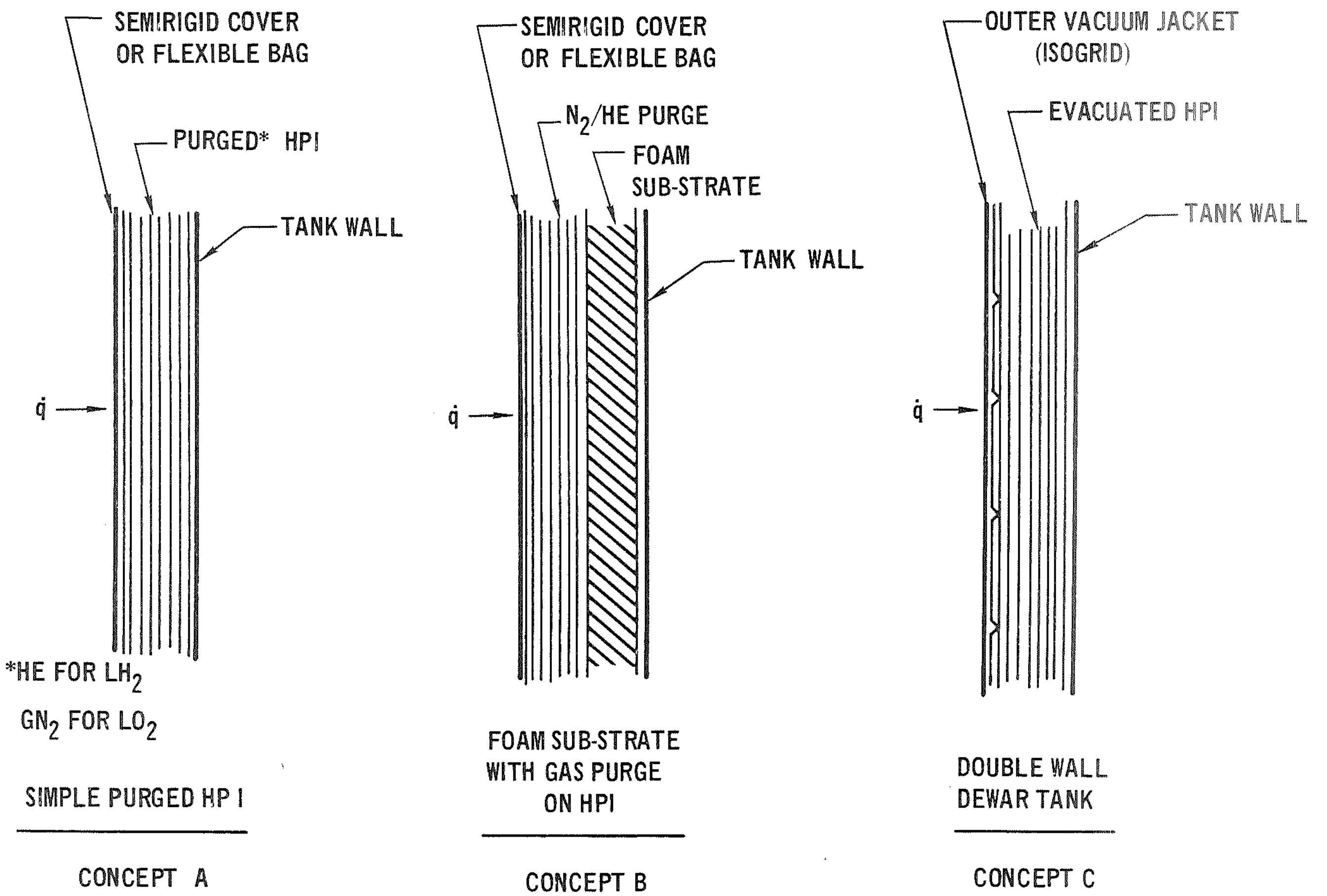
FIGURE D-7

However, the effects of multiple venting and pressurization cycles are not known. Data are available to show that unprotected insulation would freely vent during ascent without significant pressure gradients. However, no data are available to show the effects of reentry on HPI performance. It is known, though, that any form of condensation within the Mylar layers can cause severe degradation in thermal characteristics. (Condensing and freezing water within the layers can remove the aluminum coating from the Mylar.) Thus, a means of protecting the insulation from atmospheric contamination is required. The following paragraphs describe the candidate insulation concepts, their performance and the selection of the preferred concept.

Three basic insulation concepts (illustrated in Figure D-8) were evaluated for the hydrogen tank. The simplest concept uses an insulation purge with a noncondensable gas. This allows use of a semi-rigid cover or flexible bag to protect the HPI. With this concept, the effective conductivity of the insulation approaches that of the purge gas during nonvented conditions. The second approach is a substrate concept, in which foam insulation is used under HPI. The foam has a lower conductivity than helium-purged HPI and its thickness is sized so that nitrogen can be used for ascent purging rather than helium (i.e., foam provides sufficient temperature differential to prevent nitrogen condensation). For reentry, the foam will be the temperature of the liquid propellant. Thus, to avoid cryo pumping on the hydrogen tank, a noncondensing helium gas purge (concept B-1) or helium purge followed by nitrogen purge (concept B-2) will be required during reentry. The third protection concept employs a double walled dewar tank, maintaining a vacuum in the insulation. This obviously results in maximum thermal performance, because the insulation is always in a vacuum and boiloff losses during boost and entry are minimized. However, the outer vacuum jacket is a significant structural element, representing a high assembly weight penalty.

Propellant losses for a typical mission were calculated for the three concepts. These are shown in Figure D-9. A simple purge system has extremely high losses. The substrate concept reduces these by a factor of nearly five but this is still greater than the dewar system, which is subject to minimum propellant loss.

Figure D-10 compares total weight penalty for the alternate approaches. The simple purge assembly is obviously noncompetitive from a weight standpoint. Other candidates are relatively close. The dewar system would eliminate the need for purging and is competitive from a weight standpoint. However, the outer jacket



BOOST/REENTRY THERMAL PROTECTION CONCEPTS

FIGURE D-8

weights shown for this concept are probably optimistic, as they are based on use of an isogrid structure with a design safety factor of 1.5. If a safety factor of 2 were used, as assumed for the rest of the assembly design, jacket weight would increase to 360 lb. On this basis, the preferred approach for hydrogen is purged foam substrate. Although the weight penalty could be reduced by using a dual reentry purge (helium followed by nitrogen), potential weight advantage is small. For these reasons a single purge gas (nitrogen for ascent and helium for reentry) was selected. For the oxygen assembly, gaseous nitrogen can be used for purge without a substrate. Performance of candidate oxygen concepts is shown in Figure D-11. Weight penalties are smaller, and there is generally less observable difference between concepts than was true for hydrogen systems. Either a dewar tank or a simple nitrogen purge system appears practical. As with the hydrogen systems discussed above, the dewar weights are believed to be optimistic. Therefore the nitrogen purge system was selected for oxygen.

	GROUND HOLD	BOOST	EVACUATION	ORBIT	RE-ENTRY	TOTAL	REMARKS
CONCEPT A	17.7	161	8.0	71	0.3	258	
CONCEPT B							
B-1 N_2 PURGE BOOST He ENTRY PURGE	2.6	19.1	3.3	71	0.3	97	0.42 IN. FOAM
B-2 N_2 PURGE BOOST He/N_2 DUAL GAS ENTRY PURGE	2.6	19.1	3.3	71	0.3	97	0.42 IN. FOAM
CONCEPT C	0.01	0.09	0.4	71	0	72	

TOTAL MISSION LH_2 LOSSES
(All Heat Input Goes Into Boiloff)

FIGURE D-9

	CONCEPT A SIMPLE He PURGE	CONCEPT B-1 FOAM SUBSTRATE SINGLE ENTRY PURGE GAS	CONCEPT B-2 FOAM SUBSTRATE DUAL ENTRY PURGE GAS	CONCEPT C DEWAR TANK
LH_2 VENT LOSS	258	97	97	72
HARDWARE WEIGHTS				
INCREASED TANKAGE	77	23	15	7
FOAM INSULATION	–	26	26	–
HPI OUTER JACKET	69 (2)	62 (2)	63 (2)	270 (3)
PURGE GAS SUPPLY SYSTEM	15 (1)	15 (1)	40	–
TOTAL	161	126	144	277
TOTAL WEIGHT PENALTY	419	223	241	349

HYDROGEN THERMAL PROTECTION SYSTEM COMPARISON
Low P_c APS

(1) IT WAS ASSUMED THAT HELIUM COULD BE TAKEN FROM EXISTING HELIUM RESIDUALS IN THE VEHICLE AND NO PENALTY WAS CHARGED FOR THE GAS OR STORAGE BOTTLE.

(2) OUTER JACKET IS 0.02 IN FIBERGLAS.

(3) ALUMINUM VACUUM JACKET USING ISOGRID STRUCTURE CONCEPT.

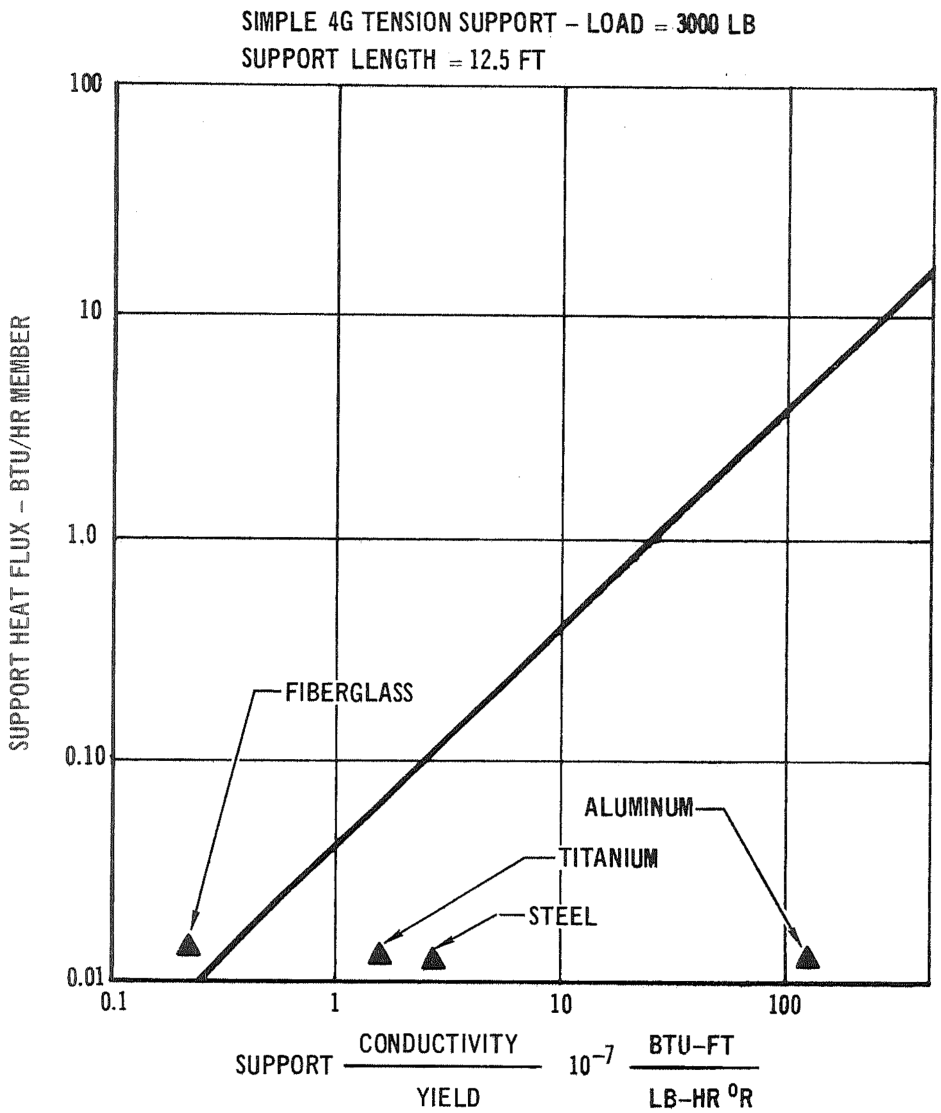
FIGURE D-10

	SIMPLE N_2 PURGE	DEWAR TANK
LO_2 VENT LOSS	48 LBS	32
HARDWARE WEIGHTS		
INCREASED TANKAGE	2	2
FOAM INSULATION	–	–
HPI JACKET	20	50
PURGE GAS SUPPLY SYSTEM	25	–
TOTAL	47	52
TOTAL WEIGHT PENALTY:	95 LBS	85

OXYGEN THERMAL PROTECTION SYSTEM COMPARISON

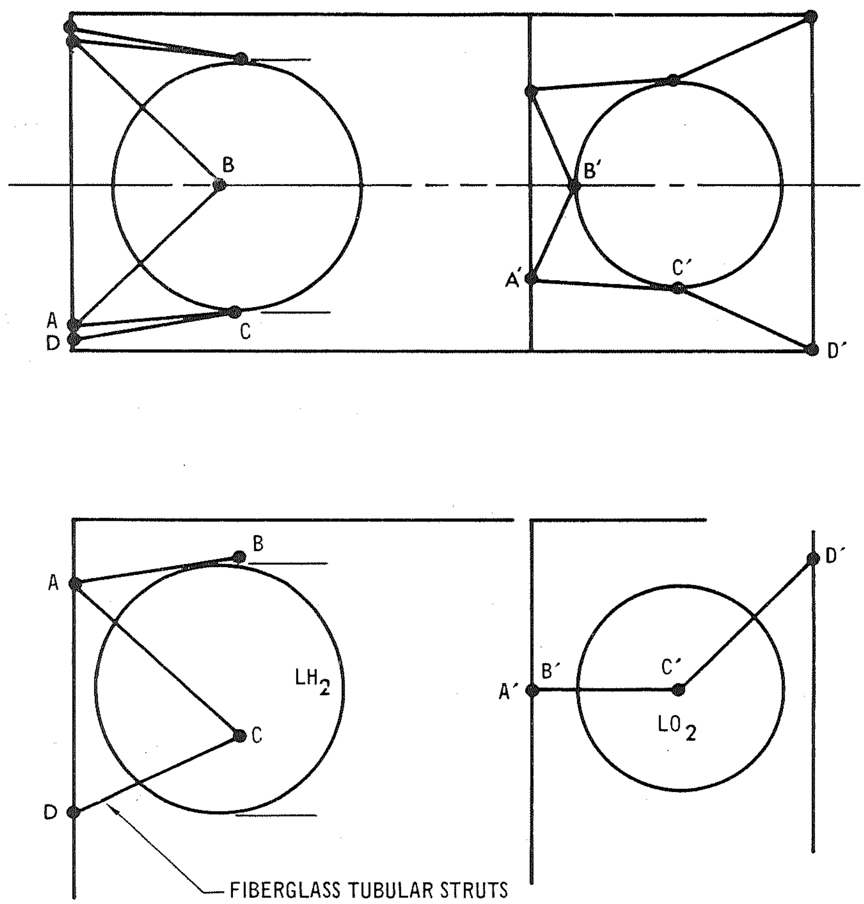
FIGURE D-11

D-3.3 Tank Heat Shorts - Multilayer HPI is used to reduce propellant heat transfer through tank wall. However, in any cryogenic tankage design, it is necessary to give careful attention to the various heat shorts associated with tank mounting structure and feed lines. Previous studies have shown that point support with low conductivity trusses must be used to provide low support losses. Figure D-12 shows the influence of support materials on general thermal performance. As shown, fiberglass is one of the most attractive materials for tank support because of its high strength, low density and low conductivity. Figure D-13 shows a typical layout for a tubular strut tank support system. Practical strut dimension, weight, and heat transfer characteristics are also shown. Analyses were conducted to identify heat short through the various lines to the tank. Results are shown in Figure D-14.



INFLUENCE OF SUPPORT MATERIAL SELECTION ON HEAT FLUX

FIGURE D-12



SUPPORT DIMENSIONS – IN

STRUT	LENGTH	DIA	THICKNESS
AB	126	6	0.055
AC	121	6	0.055
CD	115	6	0.100
A′B′	38	6	0.035
A′C′	62	6	0.055
C′D′	70	4.5	0.045

WEIGHT

- LH_2 SUPPORT SYSTEM ≈ 50 LB
- LO_2 SUPPORT SYSTEM ≈ 17 LB
- ATTACHMENTS ≈ 24 LB

HEAT INPUT

- LH_2 SUPPORT SYSTEM ≈ 1.1 $\frac{BTU}{HR}$
- LO_2 SUPPORT SYSTEM ≈ 0.82 $\frac{BTU}{HR}$

TANKAGE SUPPORT SYSTEM

Orbiter B

FIGURE D–13

			HEAT TRANSFER RATE BTU/HR			
			LINE CONDUCTION	GAS CONDUCTION	RADIATION	TOTAL
LO_2 TANK:	FILL/DRAIN	(2¼ IN DIA)	1.53	0.1	0.73	2.36
	FEED LINE	(2¼ IN DIA)	1.53	0.1	0.73	2.36
	VENT/PRESSURIZATION	(1 IN DIA)	0.68	0.02	0.06	0.76
	SCREEN VENTS	(0.25 IN DIA LINES)	0.68	–	–	0.68
						6.16
LH_2 TANK:	FILL/DRAIN	(2¼ IN DIA)	1.74	0.87	0.73	3.34
	FEED LINE	(2¼ IN DIA)	1.74	0.87	0.73	3.34
	VENT/PRESSURIZATION	(1 IN DIA)	0.78	0.17	0.06	1.01
	SCREEN VENTS	(0.25 IN DIA LINES)	0.78	0.04	–	0.82
						8.51

FEED LINE HEAT TRANSFER CHARACTERISTICS

Stainless Steel Lines Extending 1 Foot From Tankage Inner Shell

FIGURE D–14

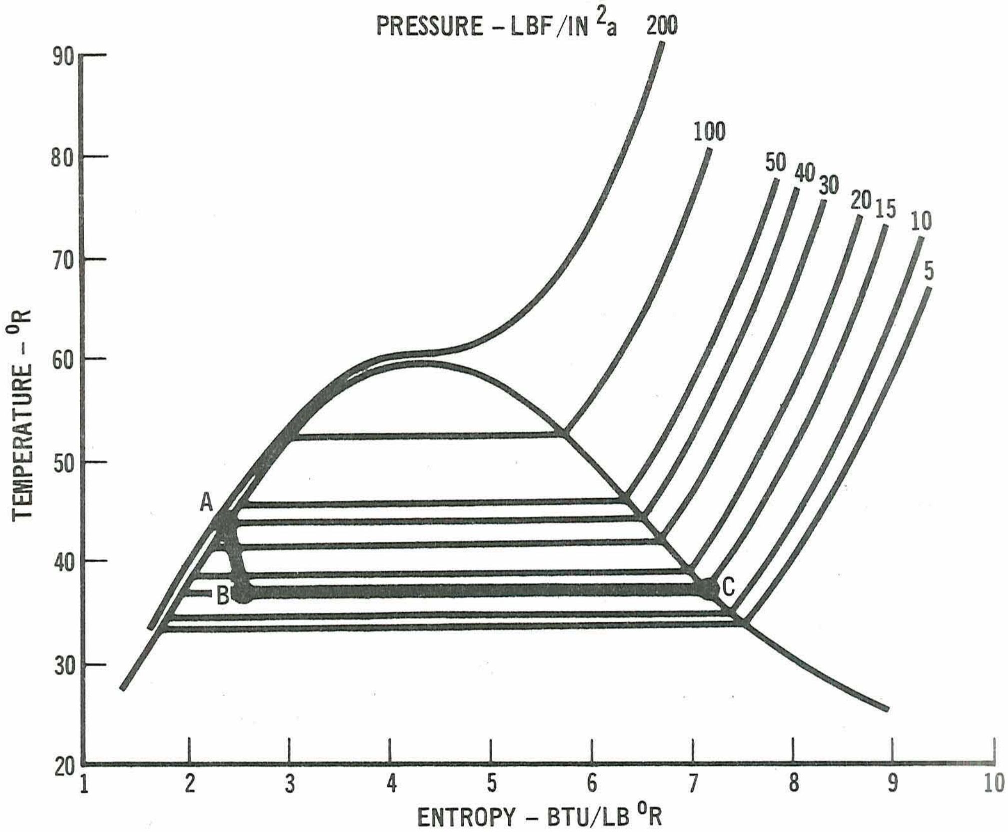
D-4. VENT/CONDITIONING ASSEMBLY

Long term storage of cryogenics requires propellant venting for tank pressure control. To avoid excessive losses, some technique for assuring vapor phase venting must be provided. One of the most promising techniques for this is the "thermodynamic separator", which converts vented liquid to gas in a heat exchanger. The operation of the device is illustrated in Figure D-15. Liquid hydrogen enters the vent system at condition A, and is throttled to point B to reduce its temperature. After throttling, a two phase mixture exists. This is circulated in a heat exchanger and heat transfer completes the vaporization until, at the heat exchanger discharge, coolant has been completely vaporized (Point C).

D-4.1 Vent System Operation - The selected vent system in which liquid hydrogen is extracted from the liquid positioning device, operates continuously during the mission. Figure D-16 is a schematic of the vent concept. There are three parallel circuits: one for feed line sump cooling, one for tank support cooling, and one for insulation, pressurization and line cooling. The first two are essentially the same for all heat exchanger approaches. All three circuits are throttled to the same pressure, so that downstream pressure remains constant. Hydrogen is extracted from the positioning device and throttled to reduce its temperature by approximately 7°R. Definition of a fixed temperature difference in this manner allows heat exchanger design to be independent of final tank pressure selection. Seven degrees R was found to give a good balance between number of coils for feed line/sump cooling and reasonable tube sizes for insulation cooling heat exchanger. Hydrogen exhausted from the hydrogen tank cooling circuit is used to provide oxygen tank cooling.

D-4.2 Heat Exchanger Concepts - The principal technical issue to be resolved in vent assembly design was definition of the heat exchanger concept to be used. Three basic options are feasible; (1) heat exchanger mounted directly to tank structural shell, (2) radiation shroud heat exchanger in which cooling tubes are separated from tank walls, and (3) compact (or internal) heat exchanger located inside the propellant tank.

The simplest heat exchanger is the wall mounted approach. This was analyzed in detail in Reference (c), where it became clear that tank wall temperature distribution could lead to a significant level of stratification which would be unacceptable to the APS, and that mixers would be required with this concept, significantly complicating design.

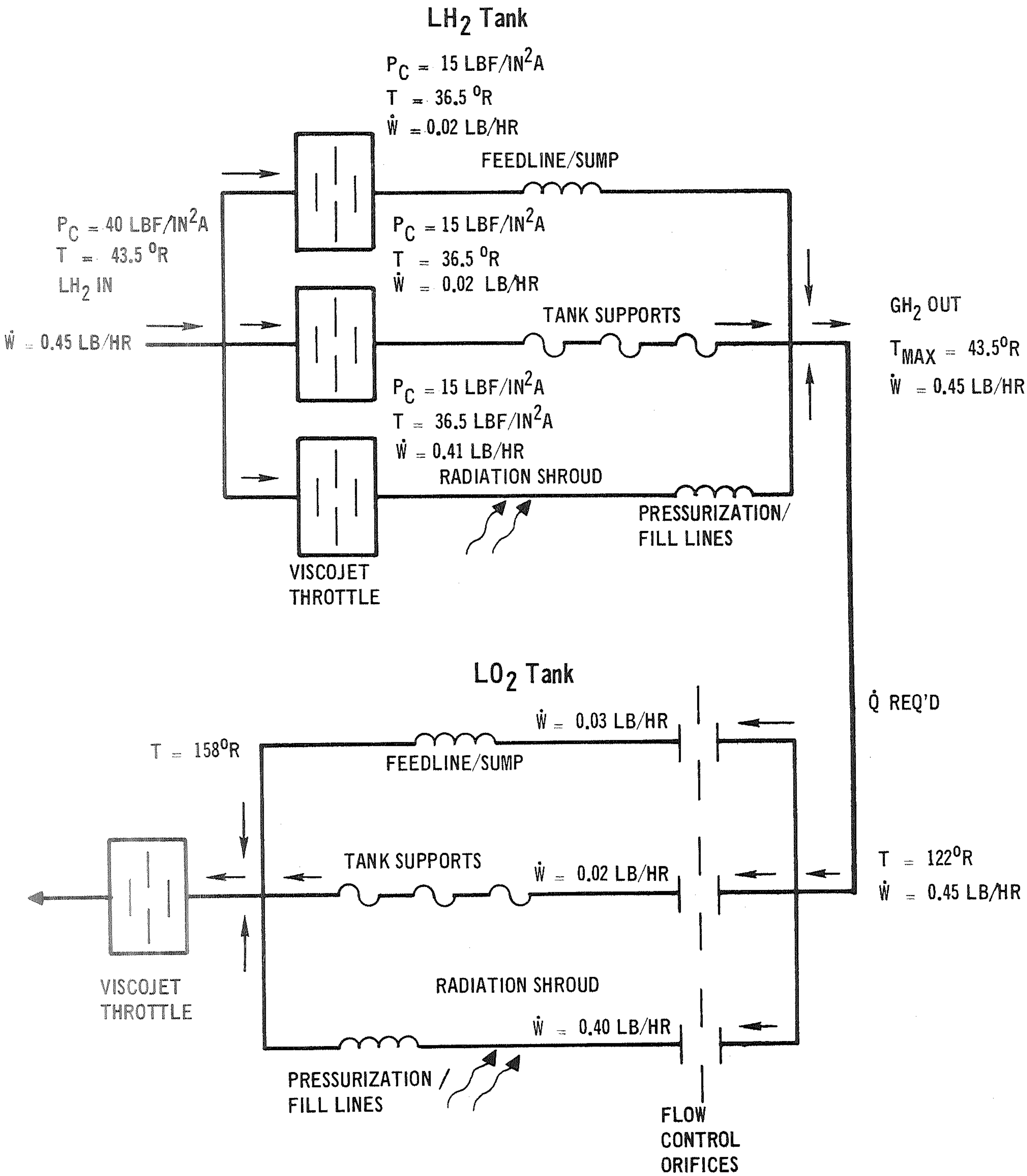


HYDROGEN THERMODYNAMIC VENT DESIGN CHARACTERISTICS

FIGURE D-15

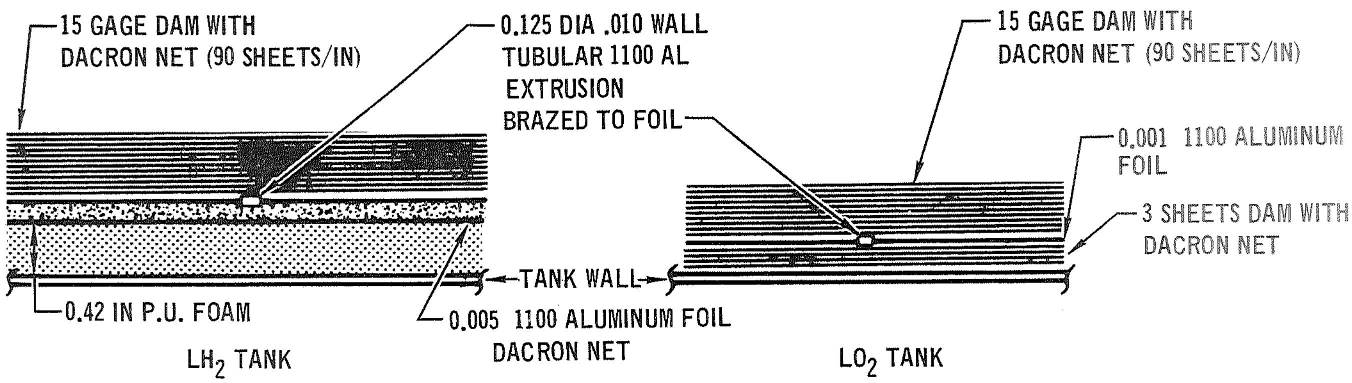
The use of a vent cooled radiation shroud, physically isolated from the tank so that radiation is the controlling mode of heat transfer, will essentially reduce radiation heat transfer to zero. Since shroud temperature is kept at or below tank temperature, all heat through the insulation can be effectively intercepted by the vaporizing coolant. Steady state performance of a device such as this was analyzed assuming the tube-shroud arrangement shown in Figure D-17.

Use of a compact, pump driven heat exchanger inside the tank has been studied and ground tested (References (d) and (e)). In this concept, tank fluid is circulated over or through the heat exchanger by a low power pump/mixer, which eliminates stratification and hot or cold spots in the tank due to unequal heat transfer. A generalized mixer sizing analysis is shown in Reference (f) and (g). Based on a conservative acceleration level of 10^{-5}g for the orbital gravity field, necessary fluid and mixer parameters were developed (Figure D-18) together with heat exchanger data.



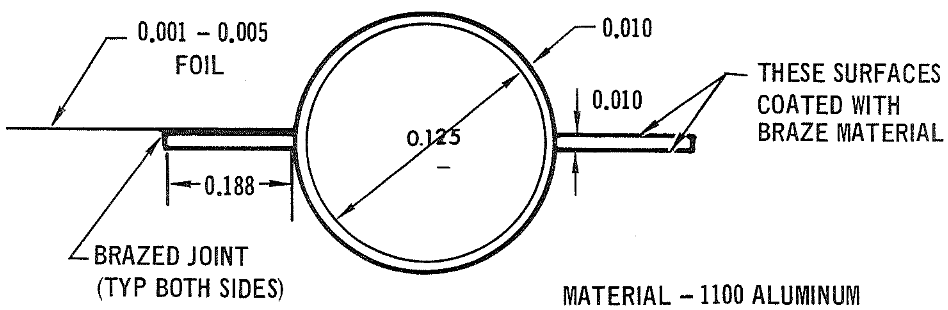
VENT SYSTEM SCHEMATIC

FIGURE D-16



COOLED RADIATION SHROUD INSTALLATION DETAIL

	D_0	PASSES	THICKNESS	WEIGHT
H_2 SHROUD	2.22 FT	14	0.005 IN	24.2 LB
O_2 SHROUD	1.92 FT	9	0.001 IN	2.3 LB



SHROUD CHARACTERISTICS

FIGURE D-17

D-4.3 <u>Heat Exchanger Concept Comparison</u> - The advantages and disadvantages of the candidate heat exchanger concepts are summarized in Figure D-19. The vapor-cooled radiation shroud concept is a light-weight, completely passive assembly, with no moving parts; therefore, it has been selected as the vent heat exchanger concept. Internal compact heat exchanger concepts have been extensively studied and have demonstrated adequate tank pressure control in 1g testing. However, no low-g tests demonstrating destratification with very low-power mixers have been performed, and the approach is complex. The tank wall-mounted heat exchanger is also a rather complicated installation, which must be integrated with the basic propellant tank. In addition, mixers would still be required.

HEAT EXCHANGER CHARACTERISTICS		
HEAT TRANSFER COEFFICIENT, BTU/HR-FT^2 – ^oR		
INSIDE		21.1
OUTSIDE		14.2
AREA, IN^2		199
TUBE DIAMETER, IN		0.25
TUBE LENGTH, IN		253
HEAT EXCHANGER WEIGHT, LB		0.6
MIXER CHARACTERISTICS		
	H_2	O_2
INPUT POWER, WATTS	1.2	5.1
EFFICIENCY, PERCENT	10.0	14.0
MIXER WEIGHT, LB	0.6	0.8
OUTLET DIA, IN	1.0	1.0
BLADE DIA, IN	1.9	1.9
INTERFACE FLUID VELOCITY,FT/SEC	0.00592	0.00685
PUMPED FLOW RATE FT^3/MIN	0.127	0.081
INCREASED H_2 VENT, LB	4.0	–

INTERNAL HEAT EXCHANGER/MIXER CHARACTERISTICS

FIGURE D–18

CONCEPT	ADVANTAGES	DISADVANTAGES
INTERNAL	1. LIGHTWEIGHT 2. STRATIFICATION ELIMINATED. 3. SIMILAR SYSTEMS ARE STATE-OF-THE ART AND HAVE BEEN GROUND TESTED. 4. LEAKAGE WILL NOT DEGRADE H_2 INSULATION 5. FAIRLY SIMPLE INSTALLATION	1. MIXER REQUIRED 2. O_2 MIXER MUST BE PURGED 3. SLIGHT ADDITIONAL HEAT TO PROPELLANT REQUIRING ADDITIONAL VENTING. 4. MIXER RELIABILITY
WALL MOUNTED	1. FAIRLY LIGHTWEIGHT 2. STRATIFICATION ELIMINATED 3. SYSTEM IS STATE-OF-THE-ART AND HAS BEEN GROUND TESTED	1. MIXER REQUIRED BECAUSE OF STRATIFICATION 2. O_2 MIXER MUST BE PURGED 3. SLIGHT ADDITIONAL HEAT TO PROPELLANT REQUIRING ADDITIONAL VENTING. 4. MIXER RELIABILITY 5. COMPLEX FABRICATION AND INSTALLATION 6. LEAKAGE MAY DEGRADE INSULATION
COOLED RADIATION SHROUD	1. ELIMINATE MIXER IN TANKS 2. SIMPLEST SYSTEM (COMPLETELY PASSIVE) 3. LOWEST VENT REQUIREMENT 4. CAN BE GROUND-TESTED	1. SYSTEM HAS NOT BEEN DEVELOPED OR GROUND TESTED 2. HEAVIER 3. SOMEWHAT COMPLEX FABRICATION AND INSTALLATION

HEAT EXCHANGER CONCEPT COMPARISON

FIGURE D–19

D-5. PRESSURIZATION

Minimum pressure of 35 lbf/in^2a is required at main tank passive heat exchanger inlet. Three pressurization concepts to satisfy this requirement for hydrogen were evaluated during Subtask B. These were: (1) cold helium pressurization, (2) autogenous pressurization using compressed main engine tank vapors, and (3) low NPSP low head rise boost pumps. Cold helium was selected for oxygen pressurization, based on earlier Subtask A results. Subtask B requirements did not warrant reevaluation of oxygen pressurization concepts. The following paragraphs describe the hydrogen pressurization comparison and selection, followed by a summary of hydrogen and oxygen concepts weights and physical characteristics.

Cold Helium - Helium is stored at 3000 lbf/in^2a in a separate tank submerged within the hydrogen tank and regulated to 35 lbf/in^2a APS tank pressure. Design of the assembly was straightforward, since the pressurization process was essentially isothermal. The tank pressure level is continuously maintained at 35 lbf/in^2a. However, during extraction, the propellant vaporization rate will not be sufficient to maintain the propellant partial pressure. In this event, the partial pressure of helium will increase during extraction. After extraction has ceased, the propellant will vaporize until equilibrium vapor pressure conditions are again satisfied, and tank pressure will increase, due to excessive helium inflow, above 35 lbf/in^2a. An evaluation of the maximum pressure to be encountered during the mission indicates that the pressure rise will not exceed 40 lbf/in^2a. This pressure level is within tankage minimum gage strength capability and below the tank vent pressure, thus no weight penalty is involved.

Autogenous - With this concept, warm propellant vapors are drawn from main engine tank and compressed to 35 lbf/in^2a. Continuous tank pressurization was maintained to satisfy APS usage requirements which could occur anytime during the mission. For autogenous pressurization, the amount of liquid-vapor heat transfer is important. Heat transfer will raise liquid temperature until the liquid (or a significant portion of it) reaches saturation temperature at tank operating pressure. In addition, heat transfer from pressurant to liquid will increase required pressurant flow rate (a significant factor for compressor design). Based on approximate heat transfer calculations and baseline mission definition, maximum pressurant flow rate required to compensate for heat loss is 1.35 ft^3/sec. Added to the maximum propellant outflow rate of 1.65 ft^3/sec, total pressurant flow rate requirement becomes 3.0 ft^3/sec. Three compressor units, with two operating simultaneously to produce the requisite design conditions, provided redundancy. Compressor design data is tabulated in Figure D-20.

Boost Pumps - Consideration was also given to tank sump mounted, low head rise boost pumps. Pump design data for a completely submerged boost pump was furnished by Pesco Products, and is shown in Figure D-21. The pump would be completely submerged, with the impeller located in the APS acquisition channel sump. The pump has low weight, and requires zero NPSH and no technology advancement. Helium repressurization was deemed necessary, not from pump inlet requirements, but from the necessity of suppressing nucleate boiling, which could affect propellant acquisition later in the mission. Prepressurization to 40 lbf/in^2 was selected to provide a margin of 0.5 lbf/in^2a above vapor pressure at end of expulsion, when tank pressure is minimum.

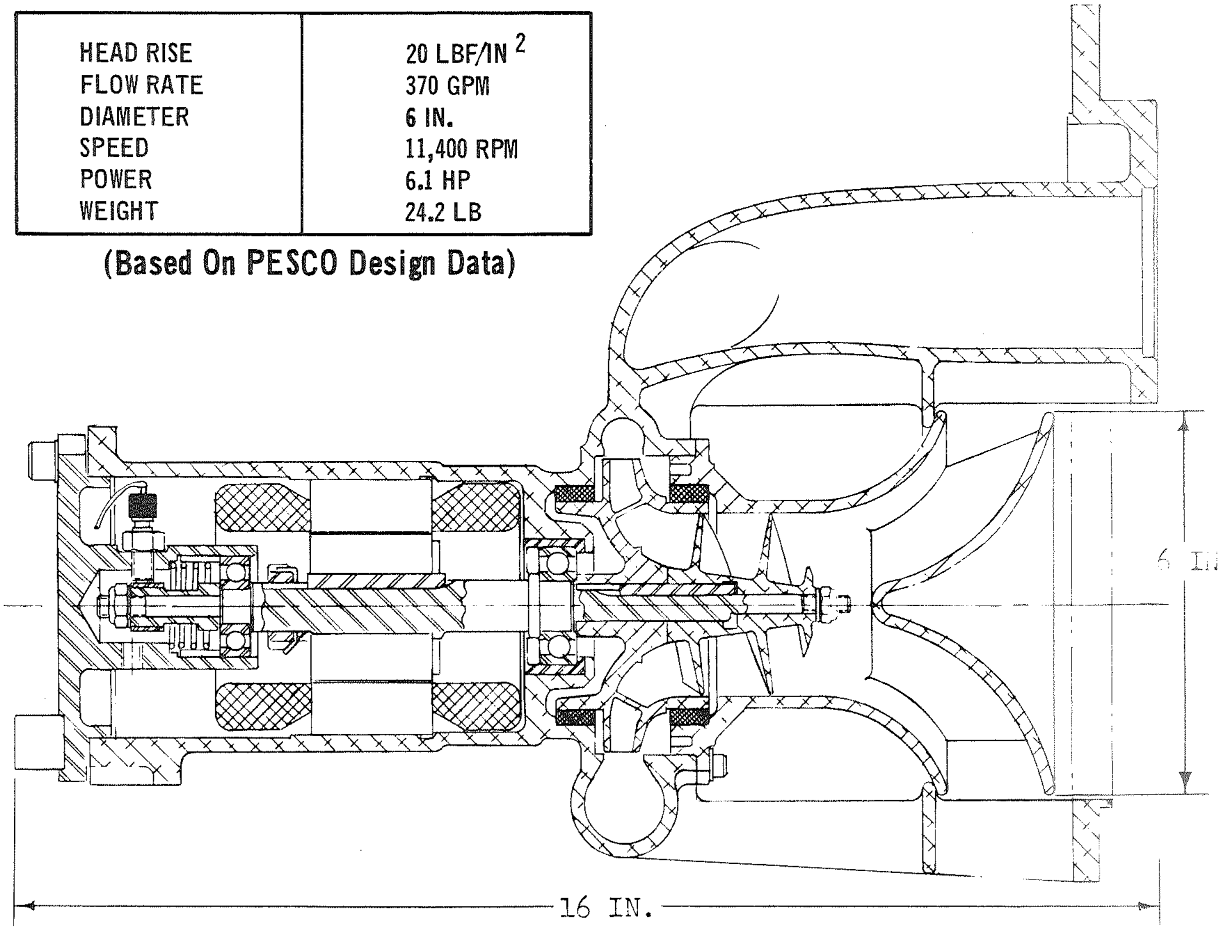
Concept Comparison - Comparison of hydrogen pressurization concepts is given in Figure D-22. Both helium and compressor approaches are heavy. The boost pump assembly was selected, therefore, on the basis of its low weight, simplicity, and desirable lack of new technology requirements.

COMPRESSOR	
INLET PRESSURE, LBF/IN^2A	20
OUTLET PRESSURE, LBF/IN^2A	30
INLET TEMPERATURE, ^0R	500
OUTLET TEMPERATURE, ^0R	641
FLOW RATE, LB/SEC	0.0126
VOLUMETRIC FLOW, CFS	1.52
SHAFT HORSEPOWER, HP	8.8
SHAFT SPEED, RPM	42,600
SPECIFIC SPEED, RPM $(CFS)^{1/2}/(FT)^{3/4}$	20
NUMBER STAGES	4
EFFICIENCY, PERCENT	43.5
HEAD RISE, FT	166,365
INTERFACE	
IMPELLER DIAMETER, IN	7.4
HOUSING DIAMETER, IN	8.9
HOUSING LENGTH, IN	10.8
INLET DIAMETER, IN	0.54
EXIT DIAMETER, IN	1.5
WEIGHT, LB	37
MOTOR	
POWER, HP	8.8
VOLTAGE	28 VDC
WATTS	8,200
EFFICIENCY, PERCENT	80
SPEED, RPM	17,000
INTERFACE	
DIAMETER, IN	8
LENGTH, IN	16
WEIGHT, LB	18
GEARBOX	
WEIGHT, LBS	15
LENGTH, IN	3
GEARBOX RATIO	2.4:1
SYSTEM	
LENGTH, IN	29
WEIGHT, LB	68

HYDROGEN CENTRIFUGAL COMPRESSOR DESIGN DATA FIGURE D-20

HEAD RISE	20 LBF/IN ^{2}
FLOW RATE	370 GPM
DIAMETER	6 IN.
SPEED	11,400 RPM
POWER	6.1 HP
WEIGHT	24.2 LB

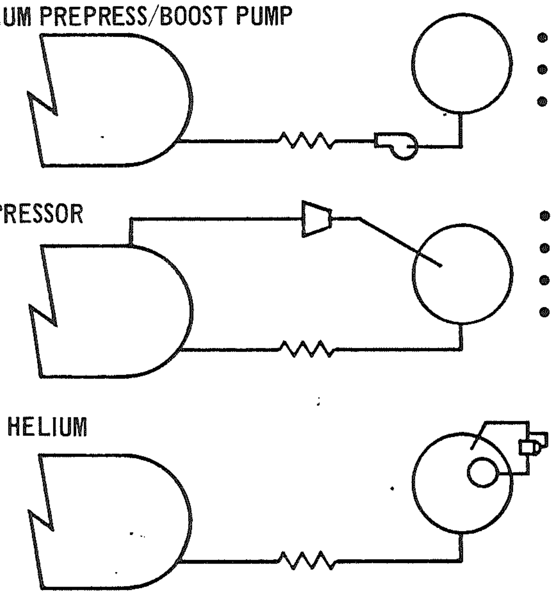
(Based On PESCO Design Data)



HYDROGEN BOOST PUMP DESIGN AND CHARACTERISTICS

FIGURE D-21

- HEAT EXCHANGER INLET PRESSURE = 35 LBF/IN^{2}A
- WEIGHTS INCLUDE REDUNDANCY



HELIUM PREPRESS/BOOST PUMP

- WEIGHT PENALTY = 170 LB
- SMALL CENTRIFUGAL PUMP √
- 6 HP ELECTRIC MOTOR

COMPRESSOR

- WEIGHT PENALTY = 333 LB
- FOUR STAGE CENTRIFUGAL COMPRESSOR
- 9 HP ELECTRIC MOTOR
- SENSITIVE TO MAIN TANK PRESSURE AND TEMPERATURE FLUCTUATIONS

COLD HELIUM

- WEIGHT PENALTY = 444 LB
- NO SIGNIFICANT POWER REQUIREMENTS

√ SELECTED SYSTEM

HYDROGEN TANKAGE PRESSURIZATION CONCEPTS

FIGURE D-22

D-6. TANKAGE INTEGRATION

APS propellants may be stored in separate, nonintegral tankage, or may be integrated with OMS propellant supply. Choice of integration concept depends on penalties involved with a combined subsystem designed for the most stringent combination of requirements compared with separate subsystems which could be designed specifically for a given set of requirements. In the case of the APS, the integrated concept must provide sufficient pressurant to meet the pressure levels required by the APS (35 lbf/in^2a) whereas the OMS would only require pressurization to 30 psia. Weights and requirements of separate and integrated concepts is given in Figure D-23. Several hydrogen pressurization schemes have been included to show their effect on subsystem weight. For separate subsystems, boost pump APS pressurization and autogenous OMS pressurization were selected. OMS weights shown reflect use of RL-10 engines (as discussed in Section 4.2.2). Here, OMS acquisition was assumed to be provided by APS settling burns. Integrated subsystem weights are best satisfied by using APS boost pumps, which can readily satisfy OMS flow requirements. Little weight difference between minimum weight concepts can be noted; thus, selection of separate or integrated tankage should be based on other factors.

Integrated tankage assemblies present several peculiar problems vis-a-vis separate tanks. Integrated tankage is larger, reducing passive acquisition subsystem design margins, and increasing development risk, since screen devices are sensitive to size and liquid head (i.e., tank length). In addition, integrated tankage is less suited to propellant off-loading, since the acquisition device must remain submerged during launch. Packaging volume differences between concepts are relatively small, and should prove to be of minimal concern. Separate concepts thus offer operational, weight, and developmental advantages, and, on these bases, were selected for the APS.

VARIABLE \ TANK CONCEPT		SEPARATE OMS/APS TANKS				INTEGRATED OMS/APS TANKS	
		APS		OMS			
PRESSURIZATION CONCEPT	(H_2)	COLD HELIUM	H_e PREPRESS PUMP	COLD HELIUM	AUTOGENOUS	COLD HELIUM	H_e PREPRESS APS PUMP
	(O_2)	COLD HELIUM	COLD HELIUM	COLD HELIUM	COLD HELIUM	COLD HELIUM	COLD HELIUM
TANK PRESSURES, (LBF/IN^2A)	(H_2)	36	40	30	30	36	40
	(O_2)	36	36	30	30	36	36
WEIGHTS, (LB)							
PROPELLANT	(H_2)	2,260	2,260	4,022	4,022	6,206	6,206
	(O_2)	5,799	5,799	18,943	18,943	24,742	24,742
TANKAGE	(H_2)	551	551	805	805	1,070	1,100
	(O_2)	236	236	520	520	680	680
THERMAL VENT SYSTEM		32	32	73	73	89	89
HELIUM SYSTEM		420	104	592	62	1,255	380
PUMP/MOTORS		–	73	–	–	–	77
ELECTRICAL POWER		–	1	–	–	–	.5
(TOTAL)		(9,298)	(9,054) (√)	(24,955)	(24,425) (√)	(34,042)	(33,275)
SUMMARY		(√) WEIGHT SEPARATE SYSTEMS 33,479 WEIGHT INTEGRATED SYSTEMS = 33,275					

(√) SELECTED

TANKAGE INTEGRATION WEIGHT SUMMARY

FIGURE D-23

D-7. TANKAGE DESIGN SUMMARY

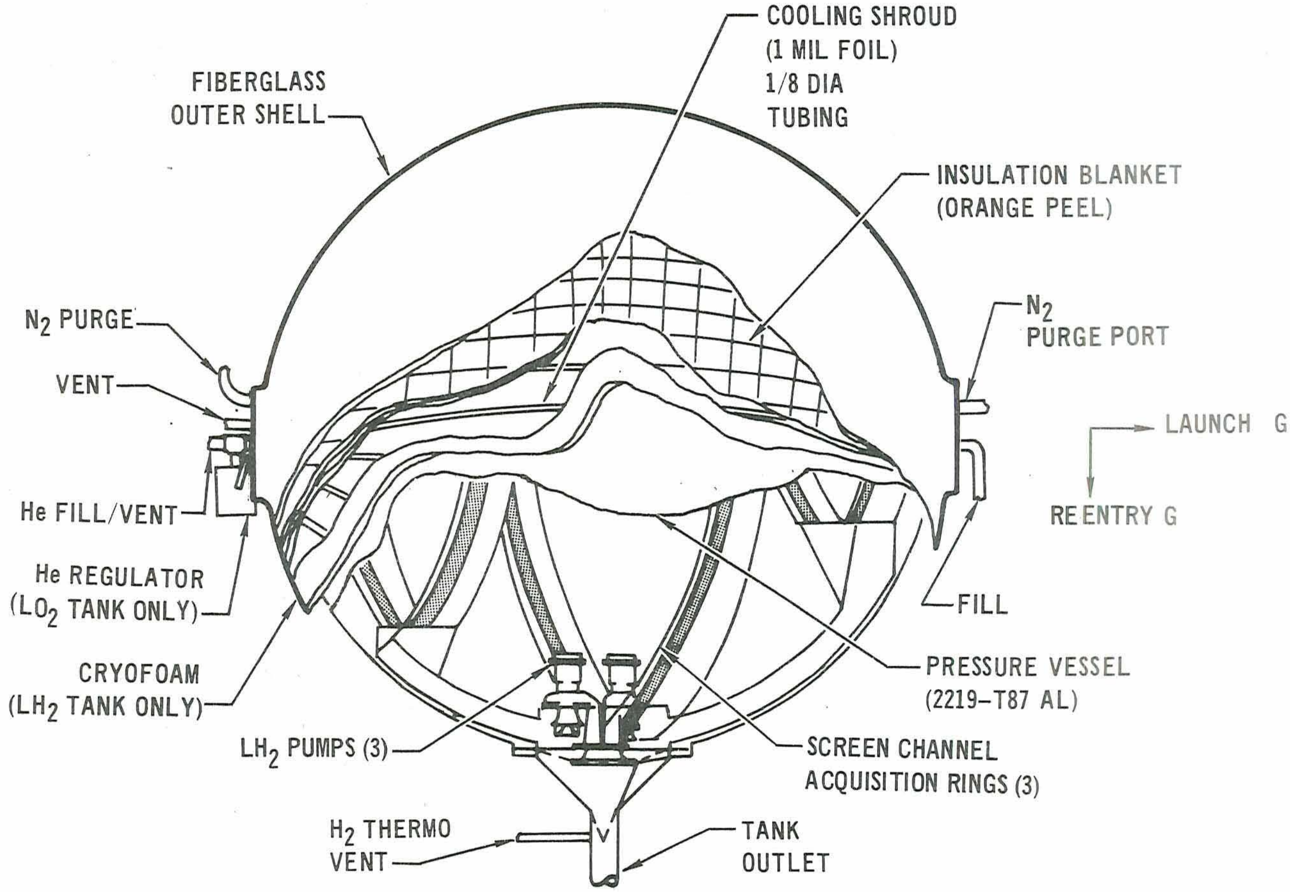
Individual subassembly design characteristics have been discussed in preceding sections. The complete assembly is summarized below.

The Orbiter LO_2 tank is pressurized by a regulated supply of helium. The helium pressurant storage tank is mounted inside the LO_2 tank to take advantage of volumetric efficiency gained by storing the pressurant at LO_2 temperatures. Hydrogen pressure is provided by low head rise boost pumps located in the tank sump. The LH_2 tank will be prepressurized with helium to 40 lbf/in^2a. As the LH_2 supply is depleted, tank pressure will drop to approximately 20 lbf/in^2a. This pressure is sufficient to maintain a positive NPSP of at least 0.5 lbf/in^2a at pump inlets, and will suppress nucleate boiling (which could be encountered during operation at saturated conditions).

The tankage concept consists of 2219 aluminum basic pressure vessel, layer of foam (on the LH_2 tank only), cooling shroud made of 0.125 inch diameter aluminum tubing brazed to an aluminum heat barrier, blanket of HPI, and fiberglass outer shell. Figure D-24 illustrates the tank assembly. Cooling shroud temperature is maintained by continuous hydrogen vent. LH_2 is extracted for cooling, expanded and subcooled 7°R, then routed to shroud, tank supports and penetrations, where it vaporizes and absorbs tank heat leak. Thermal vent requirements are shown in Figure D-25. The fiberglass outer shell serves as an environmental shield for the tank thermal insulation system. On the ground, constant GN_2 purge provides an inert atmosphere surrounding the HPI, protecting it against contamination and corrosion. On orbit, the fiberglass shell is vented to vacuum to enable HPI to function as evacuated radiation shields. On reentry either helium (for the hydrogen tank) or nitrogen (for the oxygen tank) is purged through the cavity between outer shell and tank to prevent the shell from collapsing, as well as to inhibit atmospheric contamination of the HPI.

Propellant acquisition is accomplished through use of screen channels illustrated in Figure D-26. Screen channels are placed in such a position that some portion of screen will always be "wetted", thereby ensuring fluid flow to fill tank sump channels.

A weight breakdown of the tank assemblies is summarized in Figure D-27.



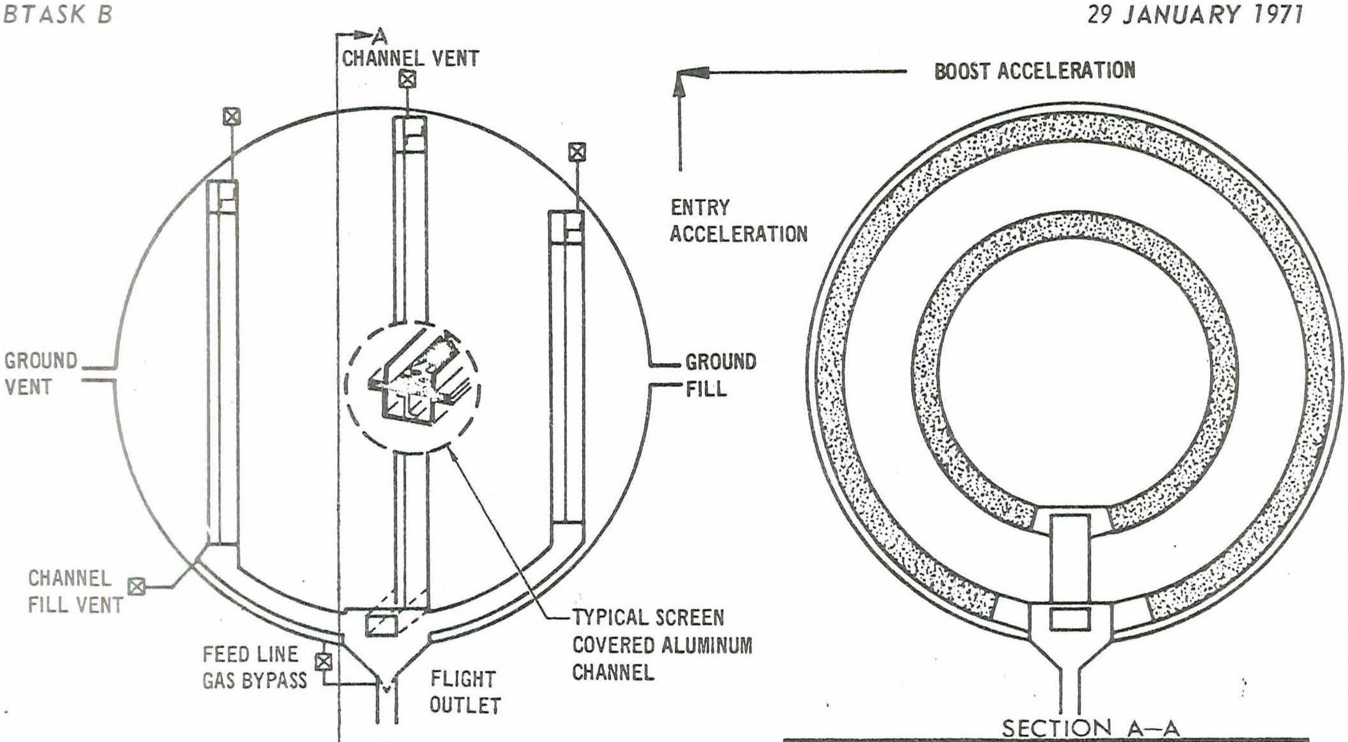
PROPELLANT TANK INSULATION/COOLING CONCEPT

FIGURE D–24

		ORBITER B	
		LH_2	LO_2
INSULATION	(BTU/HR)	72	31.3
TANK SUPPORTS	(BTU/HR)	1.1	1.0
PRESSURIZATION/FILL LINES	(BTU/HR)	5.2	3.8
SUMP/FEEDLINE	(BTU/HR)	3.4	2.4
REQUIRED VENT RATE	(LB/HR)	0.44	0
TOTAL VENTED WEIGHT (8-DAY MISSION)	(LB)	84	0

APS PROPELLANT STORAGE AND VENT SUBSYSTEM IN-ORBIT HEAT INPUT SUMMARY

FIGURE D–25



PROPELLANT	SCREEN SIZE	CHANNEL SIZE (IN)
O_2	250 x 1370 (17μ) 200 x 600 (40μ)	3 x 6
H_2	325 x 2300 (10μ) 200 x 600 (40μ)	5 x 8

APS PROPELLANT ACQUISITION DEVICE

FIGURE D–26

	HYDROGEN (LB)	OXYGEN (LB)
STRUCTURAL COMPONENTS		
PROPELLANT TANK	224	51
FIBERGLASS SHELL (INCLUDING SUPPORTS)	98	22
PROPELLANT ACQUISITION DEVICE	105	37
TANK MOUNTS	62	50
TANK INSULATION		
FOAM SUBSTRATE	31	–
INSULATION BLANKET	99	32
INSULATION SUPPORTS	10	3
COOLANT LOOP		
SHROUD	28	2
VALVES, CONTROLS	4	4
PRESSURIZATION	159	32
TOTAL	820	233

PROPELLANT STORAGE ASSEMBLY – TANK WEIGHT SUMMARY

FIGURE D–27

D-8. REFERENCES

(a) Kendall, A. S., McKee, H. B., Orton, G. F., "Low Pressure Auxiliary Propulsion Subsystem Definition", Subtask A Report, McDonnell Douglas Report No. MDC E0303, dated 29 January 1971.

(b) Orton, G. F., "Simulated Flight Vibration Testing of a Surface Tension Propellant Expulsion Screen", presented at the 12th JANAF Liquid Propulsion Meeting, 17 Nov 1970.

(c) Cady, E. C., "A Comparison of Low-G Thermodynamic Venting Systems", MDAC Report MDAC-63174, April 1969.

(d) Stark, J. A. and Blatt, M. H., "Cryogenic Zero-Gravity Prototype Vent System," Convair/GDC Report GDC-DD367-006, October 1969.

(e) Sterbentz, W. H., "Liquid Propellant Thermal Conditioning System", NASA CR 72113, April 1967.

(f) Poth, L. J., et al. "A Study of Cryogenic Propellant Mixing Techniques", Final Report, GDC/Fort Worth Report FZA-439-1, November 1968.

(g) "Low Gravity Propellant Control Using Capillary Devices in Large Scale Cryogenic Vehicles," Convair/GDC Report GDC-DDB70-008, August 1970.

APPENDIX E DESIGN

E-1. DESIGN ANALYSES

Detailed analyses were performed:

(1) to define APS design parameters such as chamber pressure, mixture ratio, and expansion ratio

(2) to establish the subsystem operating pressures and pressure balances, and

(3) to define the subsystem feed line installation details.

This effort paralleled the operating characteristics evaluation described in Appendix B, which defined subsystem performance in terms of usable propellant weight, main engine tank temperature and pressure profiles, liquid-vapor mixer requirements and performance, and, finally, engine inlet conditions. Both studies described above are strongly interdependent, but were performed separately to reduce the number of variables in each study, thus greatly simplifying the analytical effort.

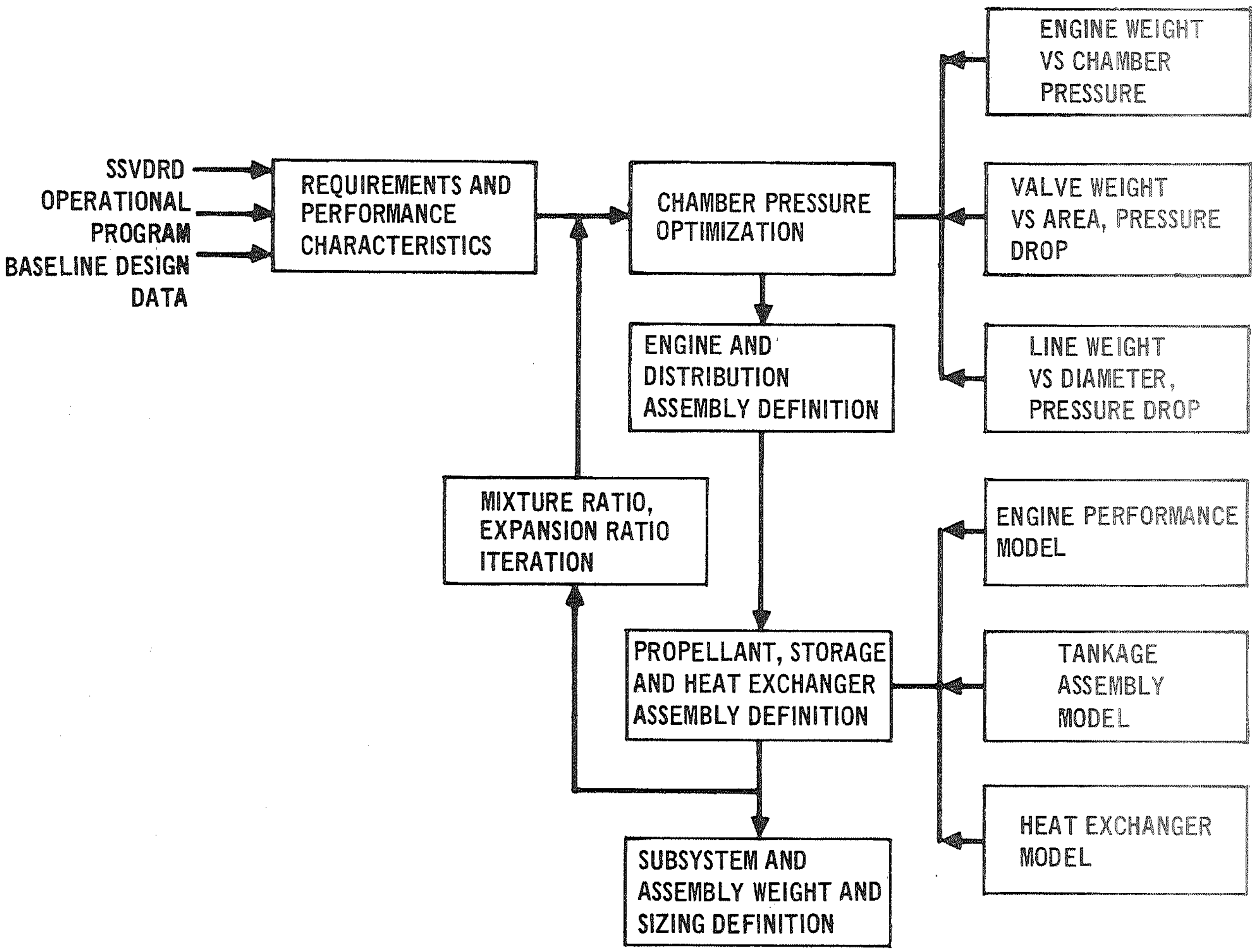
E-2. DESIGN PROGRAM

Low pressure APS inert hardware weight is a significant portion of total subsystem weight. Therefore, the APS is particularly sensitive to chamber pressure definition, feed line sizing, and subsystem mixture ratio. A computer program was developed to size the APS, and was used to define optimum subsystem design parameters. Operation of the program is illustrated by the flow diagram shown in Figure E-1.

Engines, lines, and valves were sized as a function of chamber pressure for the lowest main engine tank pressures. Each specific valve and line segment was sized for the appropriate flow rate and available pressure budget. Optimum chamber pressure was then selected to provide minimum feed system weight. Minimum weight occurs where increases in line and valve weight (due to reduced available pressure budget) are matched by decreases in engine weight (due to higher chamber pressure).

The APS upstream of the main engine tanks consists of pressurization, propellant, propellant tankage, passive heat exchanger, and liquid/vapor mixer assemblies. These were sized in accordance with models based largely on existing hardware and previous subsystem designs. These models were updated throughout the study as additional design data, such as the tankage design data in Appendix D, became available.

The sizing routines described above were performed for each expansion ratio and mixture ratio desired. In this manner subsystem weight, subassembly weights, and subassembly design characteristics were defined for each mixture and expansion ratio thus facilitating final subsystem selection.



LOW PRESSURE APS DESIGN COMPUTER MODEL

FIGURE E-1

E-3. FEED LINE DESIGN

Feed line design was established in sufficient detail to allow accurate weight estimates and chamber pressure optimization. This effort included:

(1) Feed line installation and routing

(2) Feed line sizing by segments in accordance with individual flow and pressure balances, and

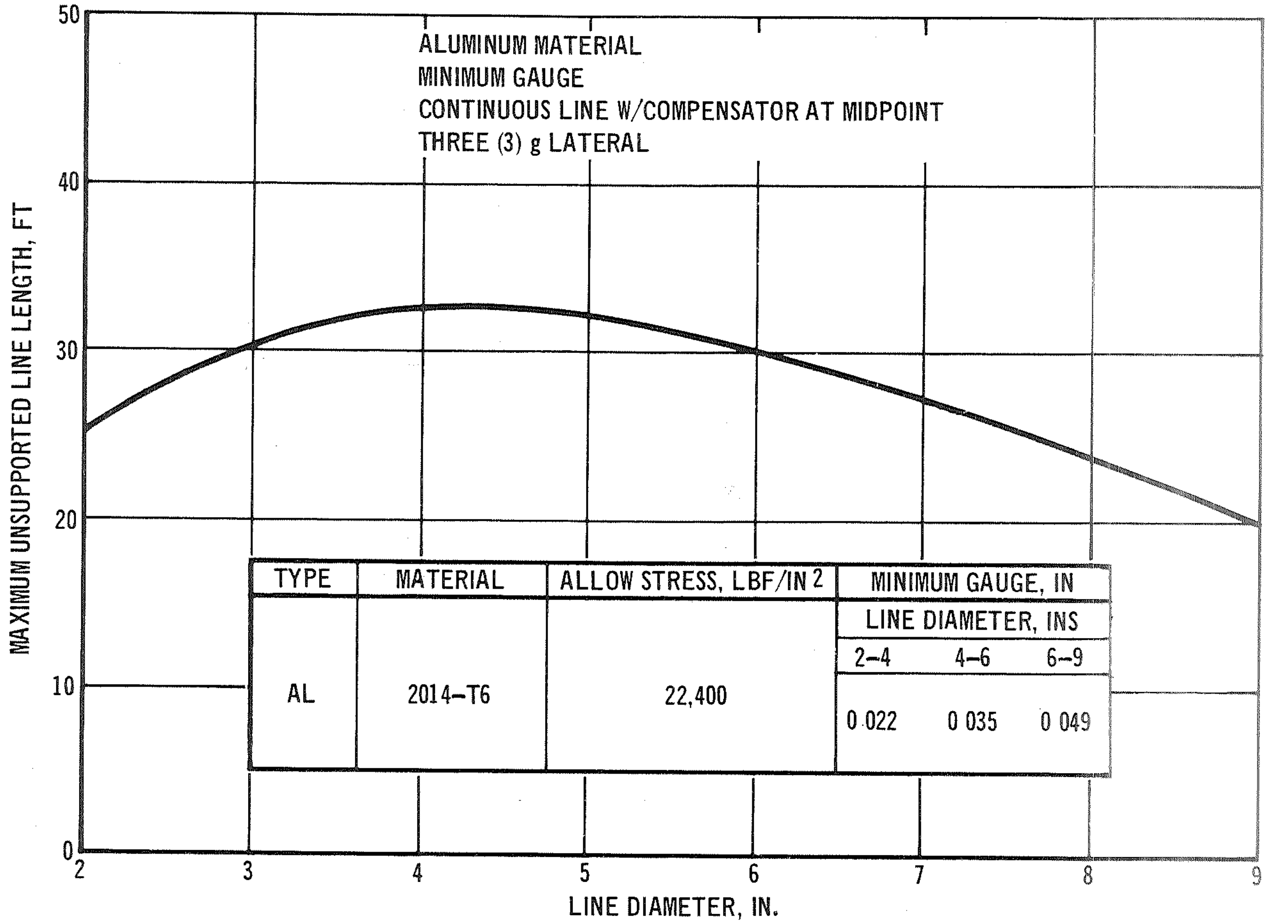
(3) Definition of line supports, joints and compensators.

The feed line distribution model, feed line sizes, and material selection were described earlier in Section 6. Specific line supports and compensator locations and sizing are described below.

Low pressure lines were sized using standard aircraft minimum gauges defined by handling and manufacturing requirements. Maximum unsupported line length was defined in accordance with induced bending stress and minimum gauge strength characteristics, and is shown in Figure E-2. Although supports are only required approximately every 30 ft for a straight run, additional supports are required in the feed system to accommodate shorter runs and manifolding.

Bellows assemblies were placed in each line segment to accommodate manufacturing tolerances, to account for large thermal contractions, and to compensate for launch vibrational motions. Both angular and linear compensators are used. A review of current compensator designs indicated that socket type angulation bellows and in-line type linear compensators provided the highest reliability, lowest pressure drop, and relatively low weights. Characteristics of these compensators are shown in Figure E-3, as are both aluminum and stainless steel weights. Weight of the stainless compensators with aluminum lines includes the use of bimetallic aluminum/stainless steel joints, where applicable. Linear compensator weights were defined using the thermal contraction associated with a 600°F temperature change (conservative) and realistic manufacturing tolerances. The compensator length was set to be three times the anticipated linear movement required, plus dia/2 for end fittings.

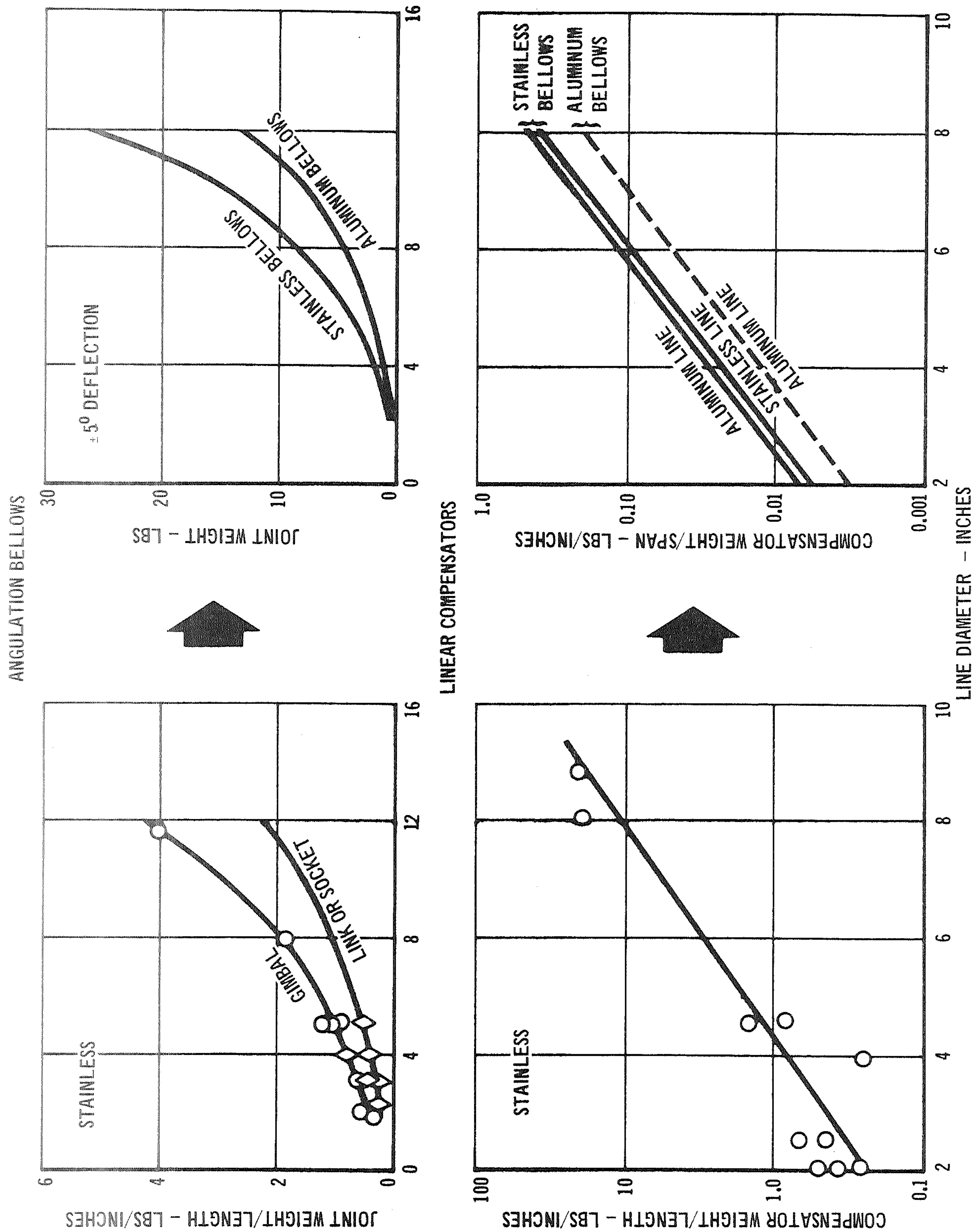
Figure E-4 presents a weight comparison of stainless steel and aluminum assemblies. A significant weight advantage is shown for the all-aluminum assembly. Though aluminum bellows have not been extensively used in the past, data from bellows manufacturers show that aluminum bellows will be satisfactory as long as high pressures and large deflections are avoided. Since these conditions do not occur in the low pressure APS, an all-aluminum assembly was selected.



TYPE	MATERIAL	ALLOW STRESS, LBF/IN^2	MINIMUM GAUGE, IN		
			LINE DIAMETER, INS		
			2–4	4–6	6–9
AL	2014–T6	22,400	0.022	0.035	0.049

MAXIMUM SPAN BETWEEN LINE SUPPORTS

FIGURE E–2



BELLOWS AND COMPENSATOR DESIGN DATA

FIGURE E-3

COMPONENT	STAINLESS STEEL BELLOWS		ALUMINUM BELLOWS	
	STAINLESS STEEL LINES	ALUMINUM LINES	STAINLESS STEEL LINES	ALUMINUM LINES
LINES	708	387	576	432
BIMETALLIC JOINTS	–	132	45	–
LINEAR COMPENSATORS	390	449	195	224
ANGULAR COMPENSATORS	287	287	144	144
TOTAL	1385	1255	960	800 (√)

(√) SELECTION

ALUMINUM AND STAINLESS STEEL LINE WEIGHT COMPARISON

Orbiter B

FIGURE E–4

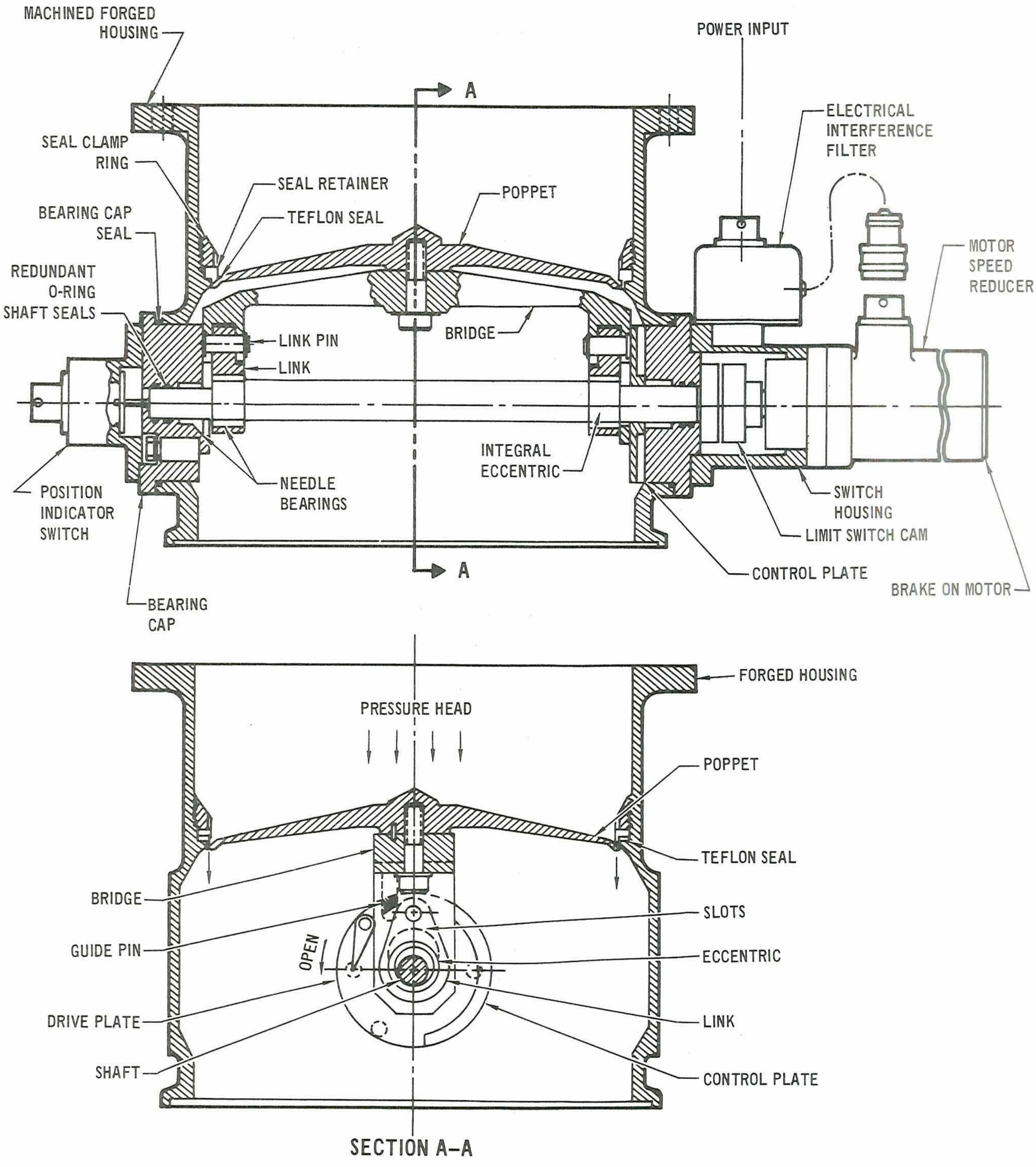
E-4. ENGINE ISOLATION VALVES

The achievement of fail operational/fail safe capability relies on isolating a failed-open or leaking engine valve, thus preventing excessive loss of propellant. Evaluation of alternate isolation valve concepts was performed during Subtask A. The selected concept uses an isolation valve for each engine ring (4 engines) to protect against a single failure, and a backup isolation valve for two rings in the event of two failures. Since the isolation valve blocks flow to four engines, actuation would result in the shutdown of operational engines. This was accommodated by providing sufficient engine thrust level and number of engines to satisfy operational (nominal) requirements with one ring inoperative and safe (minimum) requirements with two rings out. Shutdown of engine rings not only reduces the number of isolation valves required, but also prevents control axis cross-coupling after a failure, since torques remain balanced.

The isolation valve is a Martin Marietta visor-type valve, shown in Figure E-5. Valve characteristics are:

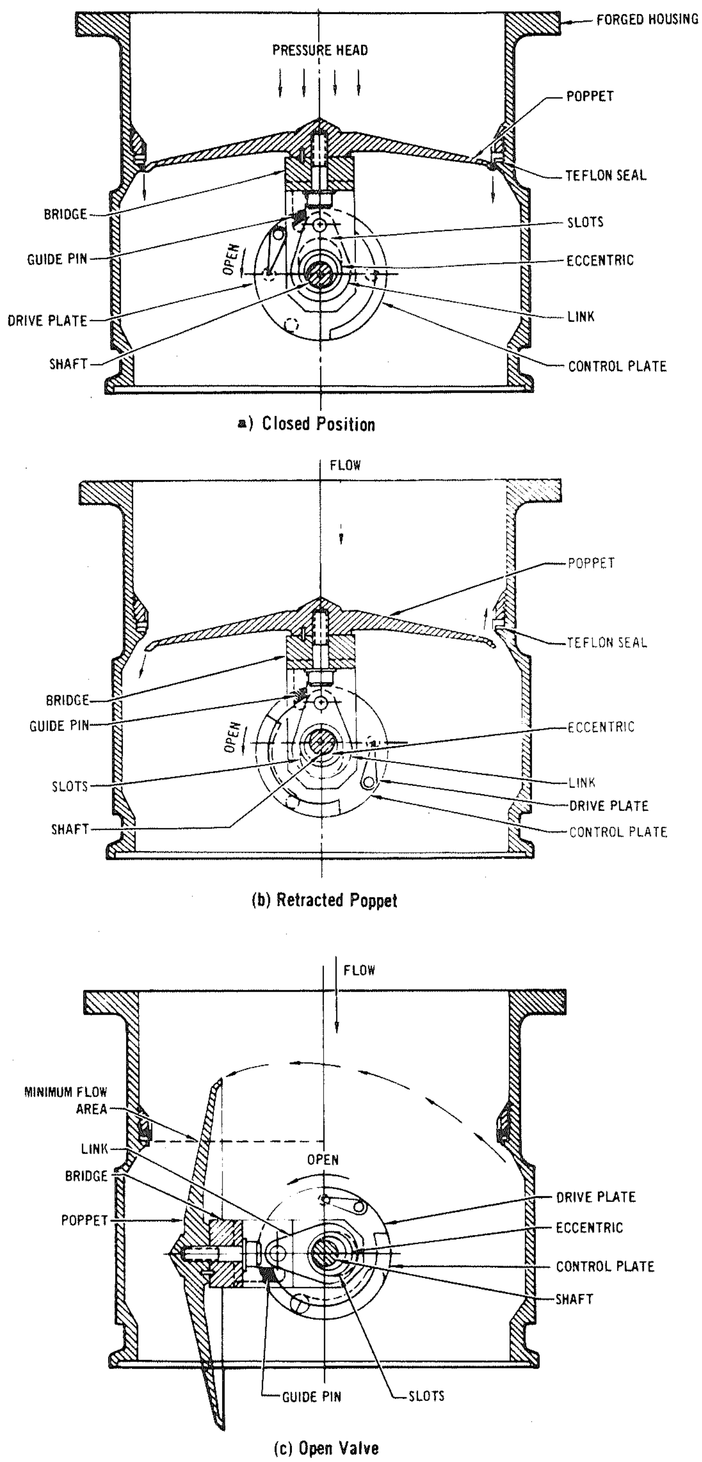
weight, lb	16
diameter, in	7
response time, sec	3.5 (0.25 sec with minor modification)
maximum leakage, SCC Helium/sec	3×10^{-6}

The valve consists of a 6061-T6 aluminum housing, 17-7PH stainless steel poppet, bridge and shaft, Teflon poppet seal, a 28 vdc permanent magnet motor and associated switches and controls. Sequence of operation is illustrated in Figure E-6. An eccentric shaft transmits a force to the bridge and poppet through a link and pin. In the initial sequence, the poppet is drawn away from the seat and held in alignment by a guide pin in longitudinal slots in the bridge. After the shaft has rotated 180 deg, the poppet and bridge are tilted to a position in line with the fluid flow. The reverse action occurs when the valve is closed. The valve is provided with position controls for both open and closed poppet. In the closed position, the poppet is locked by over center camming and by locking the shaft through the reduction gearing by a magnetic brake on the motor shaft, while in the open position, the valve shaft is locked by the same magnetic brake and the poppet is held open by fluid flow forces.



MARTIN MARIETTA CORP. ISOLATION VALVE
(Shown in Closed Position)

FIGURE E–5



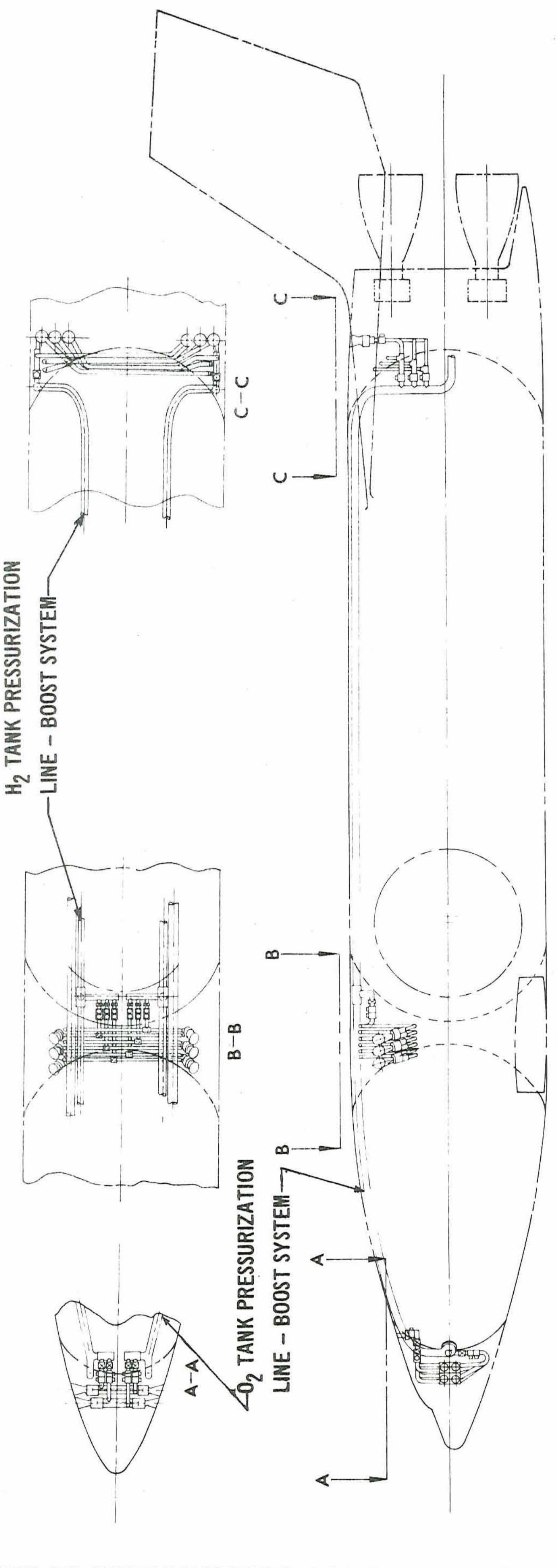
MARTIN MARIETTA CORP. ISOLATION VALVE

FIGURE E-6

E-5. APS INSTALLATION

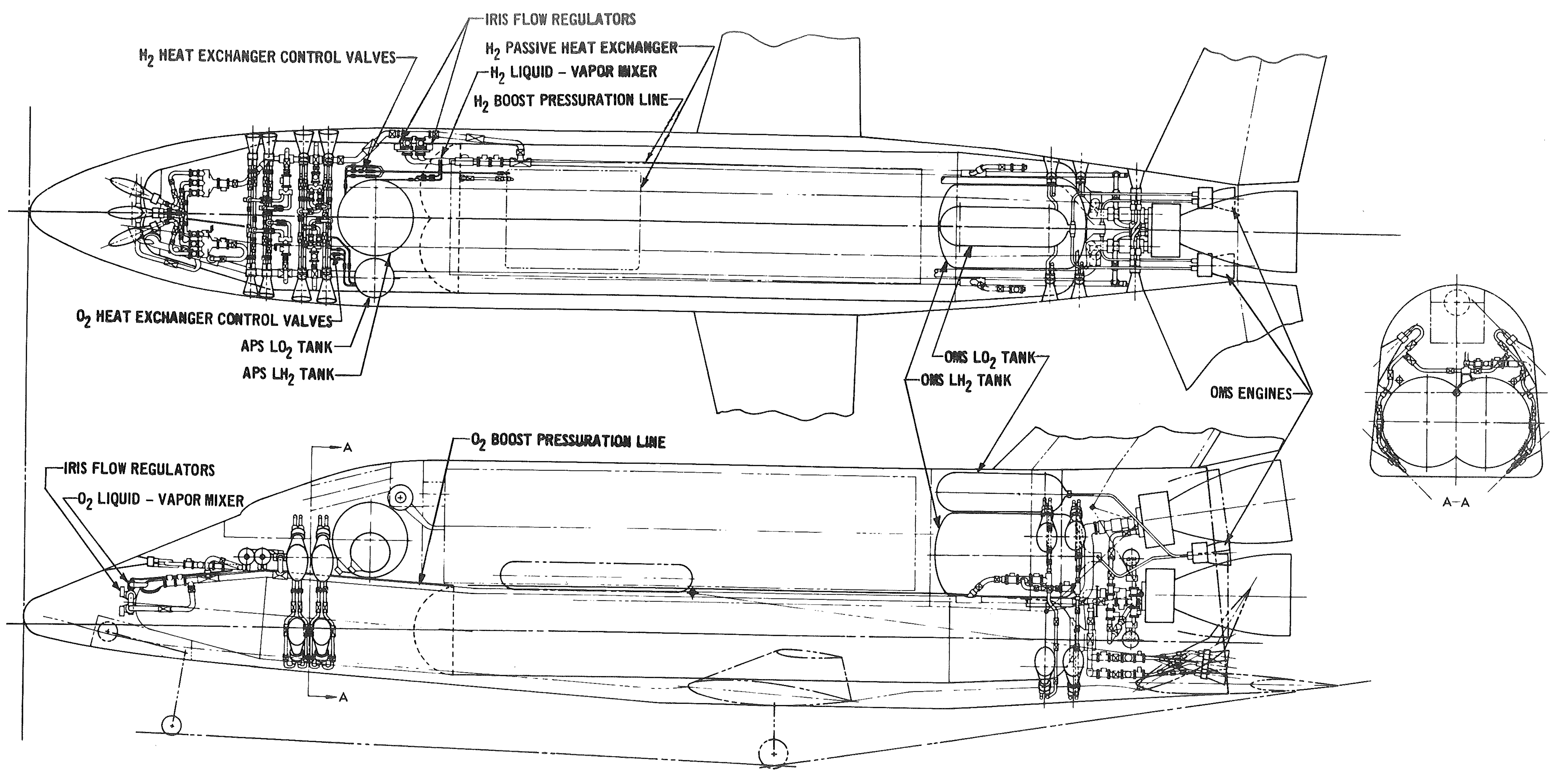
The booster APS installation is shown in Figure E-7. The engines were installed in available locations forward, between and aft of the main engine tanks. This arrangement proved to be the most advantageous in terms of APS engine number and thrust levels (Section 3.0). The subsystem uses the main engine pressurization lines for propellant distribution, thus minimizing APS manifolding. A total of twenty 2500 lb thrust engines have been installed in such a manner that no engine penetrates the lower thermal protection. The forward pitch-roll engines are canted outward to avoid APS plume impingement on the orbiter during critical staging operations.

The orbiter APS installation is shown in Figure E-8. Again the engines have been installed forward and aft of the main engine tanks, and no engines penetrate the lower heat sheild surface. Attitude and vertical maneuvering requirements are satisfied by the four rings of four engines each. Yaw and lateral translation maneuvers are accomplished with two groups of side mounted engines, installed fore and aft of the vehicle cg. Six base-mounted and three nose-mounted engines provide fore/aft translation. A separate, high thrust OMS provides the large maneuvers and is installed in the aft end of the orbiter. The OMS installation reflects propellant and engine envelopes associated with an RL10 engine subsystem providing 1150 ft/sec maneuvering capability. The APS performs the remaining small maneuver and attitude control requirements. Separate APS storage tanks have been provided forward of the cargo module to meet APS requirements. Liquid propellants from the APS storage tanks are delivered to tank mounted heat exchangers, where they are conditioned for main engine tank resupply. Vapors are drawn from the main engine tanks, mixed with liquid propellants to control engine inlet conditions, and delivered to the engines through main engine tank pressurization lines.



INSTALLATION – LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM – BOOSTER

FIGURE E-7



INSTALLATION – LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM – ORBITER

FIGURE E-8

APPENDIX F

SPACE SHUTTLE VEHICLE DESCRIPTION AND REQUIREMENTS

F-1. INTRODUCTION

This appendix describes the vehicle configurations and mission requirements which were used as a basis for the Subtask B APS design definition studies. Two vehicles, one booster and one orbiter, were selected by the NASA. The booster was a single-body stage employing swept wings and a canard. The orbiter vehicle selected (identified as orbiter B) was a straight wing low cross-range stage which was designed to reenter at a high angle of attack to minimize vehicle heating rates and temperatures. The overall vehicle/mission requirements, subsystem and component design criteria and vehicle configurations and characteristics are described in the following paragraphs. Deviations from the requirements of this appendix, as noted elsewhere in this report, were made with the expressed knowledge and consent of the NASA technical director.

F-2. MISSION REQUIREMENTS

A general description of the missions and their requirements are shown in Figure F-1. The reference mission used in designing the APS was the logistics resupply mission of a space station or space base. The design reference orbit was a 270 nautical mile circular orbit, with a 55 degree inclination.

The mission timelines for the Space Station/Base Logistics Mission are presented in Figures F-2 and F-3. Because the in-plane phasing with a space station generally will be random over long periods of shuttle launch opportunities, an early (3rd orbit) rendezvous and a late (17th orbit) rendezvous are necessary to cover both small and large phase angle variations at orbit insertion. Figure F-2 presents the orbiter third-orbit rendezvous timeline. Figure F-3 presents the 17th orbit rendezvous timeline, and Figure F-4 presents the booster timeline. The ΔV requirement to be provided for attitude maneuvers not listed in the timelines, is 10 ft/sec for the orbiter and 4 ft/sec for the booster, both based on vehicle main engine cutoff weights.

The maneuvering requirements for the orbiter and booster during the Space Station/Base Logistics Mission are shown in Figures F-5 and F-6.

Orbital Characteristics \ Missions	Space Station/ Base Logistics Support	Placement and Retrieval of Satellites	Delivery of Propulsive Stages & Payload	Delivery of Propellants	Satellite Service & Maintenance	Short Duration Orb. Mission
Altitude (n. mi.)	200 - 300	100 - 800	100 - 200	200 - 300	100 - 800	100 - 300
Inclination (deg)	28.5 - 90	28.5 - Sun Syn.	28.5 - 55	28.5 - 55	28.5 - Sun Syn.	28.5 - 90
On-Orbit ΔV (1000fps)	1 - 2	1 - 5	1 - 1.5	1 - 2	1 - 5	1 - 2
On-Orbit Stay Time (days)	7	7	7	7	7 - 15	7 - 30
Crew	2	2	2	2	2	2
Passengers (min)	Rotate 50 men/qtr	2	2	2	4	12
Discretionary Payload						
Weight (1000 lbs)	*70/qtr	— — — —	— — — —	— — — —	— — — —	— — — —
Volume (1000 ft^3)		5 - 10	10	10	5 - 10	4 - 6
Critical Dimen. Dia. (ft)	10 - 15	15	15	15	15	15

* Include Passengers

MISSION CHARACTERISTICS

FIGURE F-1

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
1. Liftoff	00:00:00	No APS requirement.
2. Staging		Damp separation rates. Provide roll control for orbiter boost engine-out condition.
3. Insertion into 50 × 100 N. mi. Orbit	00:07:34	Maintain cutoff attitude. Damp boost engine cutoff transients. Deadband ± 0.5°.
4. Insertion Orbit Determination and Prethrust Targeting		Maneuver to local horizontal, heads down, + X in direction of motion. Impart orbital rate to maintain local attitude. Deadband ± 5°.
5. Prethrust Attitude Maneuver	00:39:15	Maneuver to burn attitude and maintain inertial attitude. Hold at ±0.5° deadband.
6. Phasing Burn into 249 × 100 n. mi. Orbit	00:49:15	Horizontal, in-plane, posigrade, heads-up maneuver, 350 fps ΔV.
7. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to local horizontal, head up, + X in direction of motion, impart orbital rate. Deadband ± 5°.
8. Prethrust Attitude Maneuver	01:24:47	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5°-deadband.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE
ORBITER – THIRD ORBIT RENDEZVOUS

FIGURE F-2

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
9. Height Adjustment Burn into 260 × 249 n. mi. Orbit	01:34:47	Horizontal, in-plane, posigrade, heads-up maneuver. 279 fps ΔV.
10. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to local horizontal, heads up + X in direction of motion, impart orbital rate. Deadband ± 5°.
11. Prethrust Attitude Maneuver	02:11:45	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
12. Coelliptic Burn into 260 × 260 N. mi. Orbit	02:21:45	Horizontal, inplane, posigrade, heads-up maneuver. 26 fps ΔV.
13. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to line-of-sight attitude and maintain. Deadband ±5°.
14. Prethrust Attitude Maneuver	02:56:45	Maneuver to burn attitude and maintain inertial attitude Hold at ±0.5° deadband.
15. Dispersion Burn	03:06:45	Horizontal, in-plane maneuver. 0-25 fps ΔV.
16. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to line-of-sight attitude and maintain. Deadband ± 5°.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE
ORBITER — THIRD ORBIT RENDEZVOUS (Continued)

FIGURE F-2 (CONT.)

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
17. Prethrust Attitude Maneuver	03:40:56	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
18. Terminal Phase Initiation (TPI)	03:50:56	In-plane, posigrade, heads up, pitched up at 27°, 22 fps ΔV.
19. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to line-of-sight attitude and maintain. Deadband ± 5°.
20. Prethrust Attitude Maneuver	04:00:56	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
21. MCC-1	04:02:56	In-plane, heads up, 0-36 fps ΔV.
22. Relative Tracking of Space Station and Prethrust Targeting.		Maneuver to line-of-sight attitude and maintain. Deadband ± 5°.
23. Prethrust Attitude Maneuver	04:10:56	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
24. MCC-2	04:12:56	In-plane, heads up, 0-19 fps ΔV.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE –
ORBITER – THIRD ORBIT RENDEZVOUS (Continued)

FIGURE F-2 (CONT.)

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
25. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to line-of-sight attitude and maintain. Deadband ± 5°.
26. Prethrust Attitude Maneuver	04:23:06	Maneuver to burn attitude and maintain inertial attitude. Hold at ±0,5° deadband.
27. Braking Burns*	04:25:06	45 fps ΔV over series of range/range rate gates.
28. Station Keeping and Attitude Hold	04:33:11	0 - 10 fps multiaxis translation ΔV, 0 - 10 fps multiaxis attitude ΔV ± 0.5° deadband.
29. Docking	04:55:06	0 - 10 fps multiaxis translation ΔV, 0 -10 fps multiaxis attitude ΔV ± 0.5° deadband.
30. On Orbit		Passive mode.
31. Undocking	19:53:30	0.5 fps ΔV. Retrograde. ± 0.5° deadband.
32. Separation maneuver	19:53:30	10 fps ΔV retrograde. ± 0.5° deadband.
33. Attitude Hold		Deadband ± 20°
34. Preburn Targeting	22:33:30	Maneuver to local horizontal, heads down, impart orbital rate. Deadband ± .5°.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE
ORBITER — THIRD ORBIT RENDEZVOUS (Continued)

FIGURE F-2 (CONT.)

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
35. Prethrust Attitude Maneuver	22:43:30	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
36. Deorbit Burn	22:53:30	Retrograde, in-plane, heads down. 495 fps ΔV.
37. Entry Attitude Maneuver		Maneuver to + X-axis in-plane, pitched up above local horizontal. Deadband ± 0.5°.
38. (a) Entry	23:25:30	Maneuver as required. 25-60 fps ΔV. Deadband ± 2°
(b) Entry	23:25:30	Maneuver as required. ΔV. Deadband ±
39. (a) Completion of APS Function, High Cross-Range Orbiter		Termination attitude
(b) Completion of APS Function, Low Cross-Range Orbiter		Termination attitude

* For purposes of specific APS performance studies, the following braking burn schedule should be considered typical:

Initiation Time	ΔV, fps
04:25:06	10
04:26:46	13
04:28:01	12
04:29:31	5
04:31:11	5

SPACE STATION/BASE LOGISTICS MISSION TIMELINE
ORBITER – THIRD ORBIT RENDEZVOUS (Concluded)

FIGURE F-2 (CONCLUDED)

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
1. Liftoff	00:00:00	No APS requirement.
2. Staging	00:	Damp separation rates. Provide roll control for orbiter boost engine-out condition.
3. Insertion into 50 × 100 n. mi. Orbit.	00:07:34	Maintain cutoff attitude. Damp boost engine cutoff transients. Deadband ± 0.5°.
4. Insertion Orbit Determination and Prethrust Targeting		Maneuver to local horizontal, heads down, + X in direction of motion. Impart orbital rate to maintain local attitude. Deadband ± 5°.
5. Prethrust Attitude Maneuver	00:39:14	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
6. Phasing Burn into 123 × 100 n. mi. Orbit	00:49:14	Horizontal, in-plane, posigrade, heads-up maneuver, 130 fps ΔV.
7. Random Drift		No attitude requirement defined.
8. Relative Tracking of Space Station and Prethrust Targeting	09:19:15	Maneuver to local horizontal, impart orbital rate. Deadband ± 5°.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE, ORBITER, SEVENTEENTH ORBIT RENDEZVOUS

FIGURE F-3

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
9. Prethrust Attitude Maneuver	09:39:15	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
10. Dispersion -1	09:49:15	Horizontal, in-plane, heads-up maneuver, 0-40 fps ΔV.
11. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to local horizontal, impart orbital rate. Deadband ± 5°.
12. Prethrust Attitude Maneuver	10:01:45	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
13. Dispersion -2	10:11:45	Horizontal, in-plane, heads-up maneuver. 0-25 fps ΔV.
14. Random Drift		No attitude Requirement defined.
15. Relative Tracking of Space Station and Prethrust Targeting	21:04:20	Maneuver to local horizontal, impart orbital rate. Deadband ± 5°.
16. Prethrust Attitude Maneuver	22:04:20	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
17. Height Adjustment Burn into 260 × 123 n. mi. Orbit.	22:14:20	Horizontal, in-plane, posigrade, heads-up maneuver. 282 fps ΔV.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE,
ORBITER, SEVENTEENTH ORBIT RENDEZVOUS (Continued)

FIGURE F-3 (CONT.)

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
18. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to local horizontal, impart orbital rate. Deadband ± 5°.
19. Prethrust Attitude Maneuver	22:50:08	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
20. Coelliptic Burn into 260 × 260 n. mi. Orbit	23:00:08	Horizontal, in-plane, posigrade, heads-up maneuver, 239 fps ΔV.
21. Attitude Hold		Deadband ± 20°.
22. Relative Tracking of Space Station and Prethrust Targeting	24:14:23	Maneuver to line-of-sight attitude to space station and maintain. Deadband ± 5°.
23. Prethrust Attitude Maneuver	24:24:23	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
24. Terminal Phase Initiation (TPI)	24:34:23	In-plane, posigrade, heads up, pitched up at 27°. 22 fps ΔV.
25. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to line-of-sight attitude to space station and maintain. Deadband ± 5°.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE, ORBITER, SEVENTEENTH ORBIT RENDEZVOUS (Continued)

FIGURE F-3 (CONT.)

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
26. Prethrust Attitude Maneuver	24:44:23	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
27. MCC-1	24:46:23	In-plane, heads-up maneuver. 0-36 fps ΔV.
28. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to line-of-sight attitude to space station and maintain. Deadband ± 5°.
29. Prethrust Attitude Maneuver	24:54:23	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
30. MCC-2	24:56:23	In-plane, heads-up maneuver. 0-19 fps ΔV.
31. Relative Tracking of Space Station and Prethrust Targeting		Maneuver to line-of-sight attitude to space station and maintain. Deadband ± 5°.
32. Prethrust Attitude Maneuver	25:06:33	Maneuver to burn attitude and maintain inertial attitude. Hold at ± 0.5° deadband.
33. Braking Burns*	25:08:33	45 fps ΔV over series of range/range rate gates.
34. Stationkeeping and Attitude Hold	25:16:38	0-10 fps multiaxis translation ΔV and 0-10 fps multi-axis attitude ΔV. ± 0.5° deadband.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE, ORBITER, SEVENTEENTH ORBIT RENDEZVOUS (Continued)

FIGURE F-3 (CONT.)

Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
35. Docking	25:38:33	0-10 fps multiaxis translation ΔV and 0-10 fps multi-axis attitude ΔV. $\pm$ 0.5° deadband.
36. On Orbit		Passive
37. Undocking	67:20:03	0.5 fps ΔV. Retrograde. $\pm$ 0.5° deadband.
38. Separation Maneuver	67:20:03	10 fps ΔV. Retrograde. $\pm$ 0.5° deadband.
39. Attitude Hold		Deadband $\pm$ 20°.
40. Preburn Targeting	70:00:03	Maneuver to local horizontal, heads down, impart orbital rate. Deadband $\pm$ 5°.
41. Prethrust Attitude Maneuver	70:10:03	Maneuver to burn attitude and maintain inertial attitude. Hold at $\pm$ 0.5° deadband.
42. Deorbit Burn	70:20:03	Retrograde, in-plane, heads down. 496 fps ΔV.
43. Entry Attitude Maneuver		Maneuver to + X-axis in-plane, pitched up above local horizontal. Deadband $\pm$ 0.5°.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE, ORBITER, SEVENTEENTH ORBIT RENDEZVOUS (Continued)

FIGURE F-3 (CONT.)

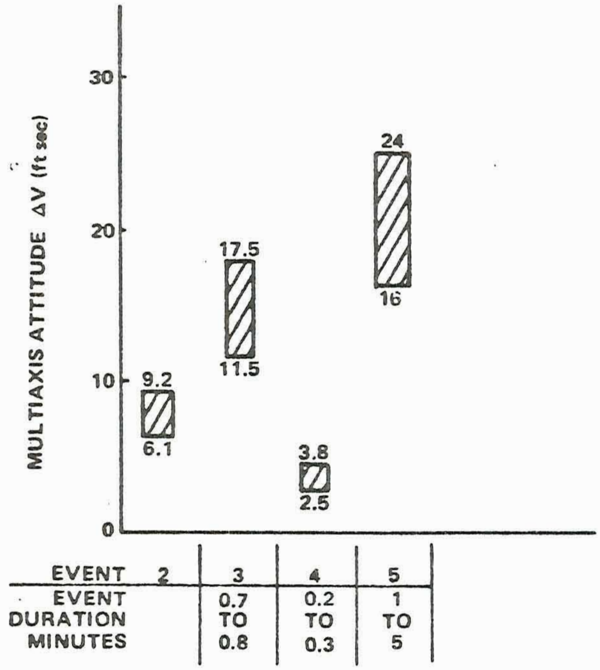
Event	Initiation Time (hr:min:sec)	Auxiliary Propulsion Requirements
44. (a) Entry	71:32:03	Maneuver as required. 25-60 fps ΔV. Deadband $\pm$ 2°.
(b) Entry	71:32:03	Maneuver as required. ΔV. Deadband $\pm$
45. (a) Completion of APS Function, High Cross-Range Orbiter		Termination attitude
(b) Completion of APS Function, Low Cross-Range Orbiter		Termination attitude

* For purposes of specific APS performance studies, the following braking burn schedule should be considered typical:

Initiation Time	ΔV, fps
25:08:33	10
25:10:13	13
25:11:28	12
25:12:58	5
25:14:38	5

SPACE STATION/BASE LOGISTICS MISSION TIMELINE, ORBITER, SEVENTEENTH ORBIT RENDEZVOUS (Continued)

FIGURE F-3 (CONCLUDED)



	EVENT COMPLETION TIME*	EVENT	PROPULSION REQUIREMENT DESCRIPTION
1.	0	Staging	Separation of booster and orbiter (No APS requirement)
2.	0+	Post Separation	Damping of main engine cutoff and separation transients.
3.	0.7-0.8	Orientation	Maneuver vehicle to reentry attitude.
4.	0.9-1.1	Attitude hold	±2° deadband
5.	1.9-6.1	Entry	±2° deadband

*Time is referenced to Event 1 in minutes unless otherwise stated. Both minimum and maximum cumulative times are shown.

SPACE STATION/BASE LOGISTICS MISSION
TIMELINE – BOOSTER

FIGURE F-4

Event*	3 Orbit Rendez.	2			3-27/32-37			28-31			38		
	17 Orbit Rendez.	2			3-33/38-43			34-37			44		
		X	Y	Z	X	Y	Z	X	Y	Z	X	Y	Z
TRANSLATION ACCELERATION ft/sec²	MIN	0.5	N/R†		0.5	0.1	0.1	0.07	0.07	0.07	N/R†		
	NOM MIN	0.65			0.65	0.2	0.2	0.1	0.1	0.1			
	NOM MAX	3.0			3.0	0.5	0.5	0.5	0.25	0.25			
	MAX	7.0			7.0	7.0	7.0	1.0	1.0	1.0			
		R**	P	Y	R	P	Y	R	P	Y	R	P	Y
ANGULAR ACCELERATION deg/sec²	MIN	0.5	N/R†		0.3	0.3	0.3	0.3	0.3	0.3	0.5	0.5	0.9
	NOM MIN	0.7			0.5	0.5	0.5	0.5	0.5	0.5	1.0	1.0	1.3
	NOM MAX	2.5			2.0	2.0	2.0	2.0	2.0	2.0	2.0	2.0	1.7
	MAX	N/R†			4.0	4.0	4.0	4.0	4.0	4.0	4.0	4.0	4.0
FINE ATTITUDE LIMITS – deg		1.0	N/R†		0.5	0.5	0.5	0.5	0.5	0.5	2.0	2.0	2.0
COARSE ATTITUDE LIMITS – deg		N/R†			20	20	20	5.0	5.0	5.0	N/R†		

* Refer to Tables I-2 and I-3 for event name corresponding to these event numbers.

** Based on roll inertia

† No Requirement

SPACE STATION/BASE LOGISTICS MISSION
ORBITER MANEUVERING CAPABILITY REQUIREMENTS

FIGURE F-5

EVENT*		2			3			4			5		
		X	Y	Z	X	Y	Z	X	Y	Z	X	Y	Z
TRANSLATION ACCELERATION ft/sec^2	MIN NOM MIN NOM MAX MAX	NO REQUIREMENT						NO REQUIREMENT					
		R	P	Y	R	P	Y	R	P	Y	R	P	Y
ANGULAR ACCELERATION deg/sec^2	MIN NOM MIN NOM MAX MAX	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0
ANGULAR RATE deg/sec	MIN NOM MIN NOM MAX MAX				2	2	2						
ATTITUDE LIMITS — deg		2	2	2	2	2	2	2	2	2	2	2	2

* Refer to Table I-4 for Event names corresponding to these numbers.

SPACE STATION/BASE LOGISTICS MISSION — BOOSTER MANEUVERING CAPABILITY REQUIREMENTS

FIGURE F-6

F-3. SUBSYSTEM/COMPONENT DESIGN CRITERIA

The APS is constrained to operate for a minimum of 100 mission cycles over an eight year period without major overhaul or refurbishment. Additionally, the APS must fail operational after failure of any criterial component, and after the second failure must provide safe operation for crew survival. Specific criteria applied to APS design are summarized below:

(1) A material ultimate factor of 2.0 was applied to stresses resulting from pressure conditions, separate from other loads.

(2) No propellant reserve was added to that estimated to be consumed in a mission cycle.

(3) Fluid line sizes were selected on the basis of design velocities that resulted in acceptable compromise between the excessive pressure drop produced by small diameter tubing and the weight and cost of large diameter tubing. Gas velocities were limited to Mach 0.3 or less.

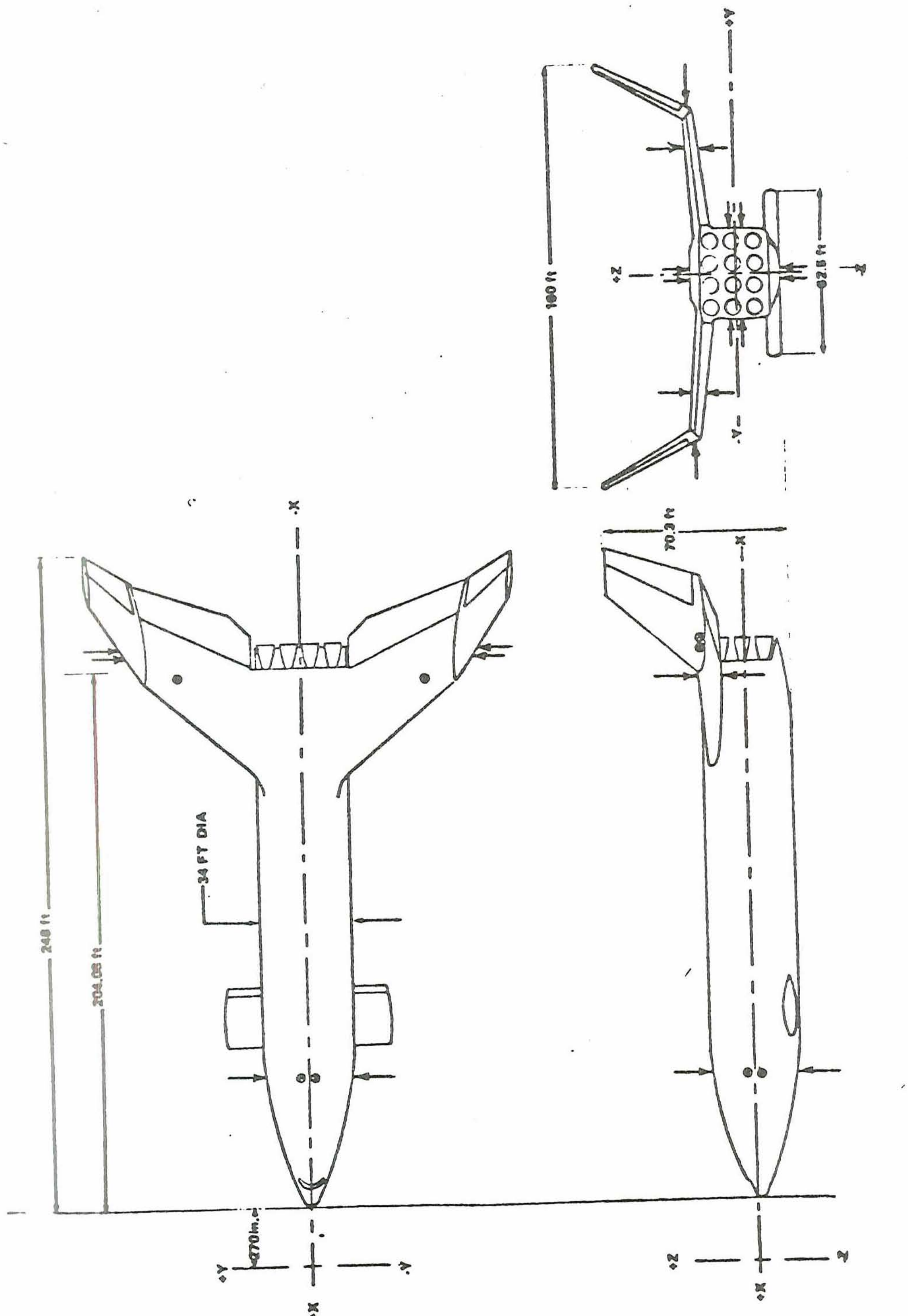
(4) Flight components were limited to those required for flight operations, except for components required for on-board checkout and servicing. Component integration, packaging, and simplicity of checkout was considered where advantages in maintainability, serviceability, replaceability, weight and cost could be realized.

(5) Provisions for ease of inspection, servicing and maintenance was incorporated in the subsystems and component design to meet the short turn-around and launch preparation requirement.

F-4. BOOSTER CHARACTERISTICS

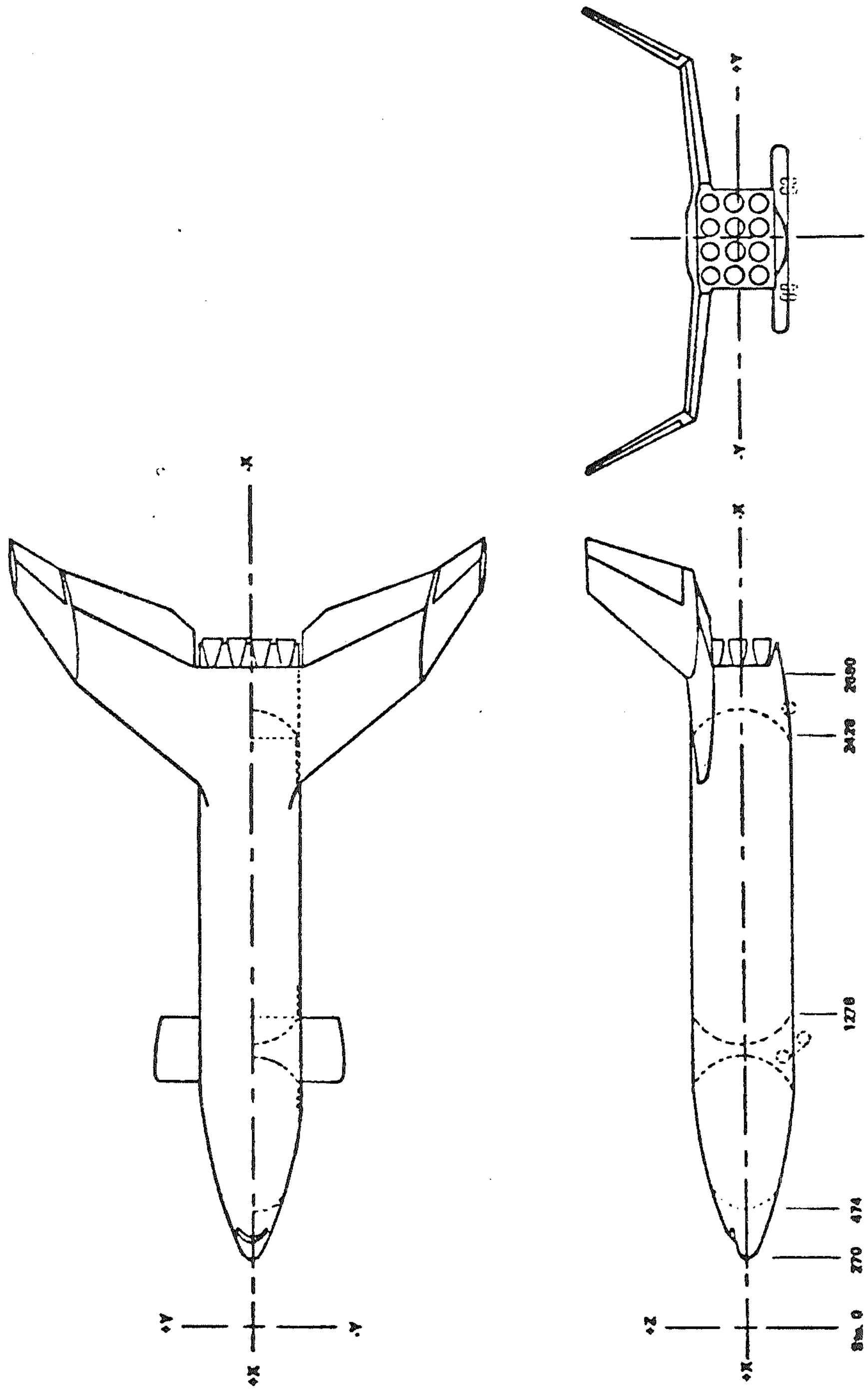
The booster is a single-body stage employing swept wings and a canard. Lift-off weight of the booster is approximately 2,840,000 lb. Airbreathing cruise engines are mounted in the canard for flyback and ferry operations.

External and internal profile information is presented in Figures F-7 and F-8. Further information on mass characteristics, tankage, propellant residuals, and heating environments is given in Figures F-9 through F-11. Auxiliary propulsion thruster locations are shown in Figure F-12. The relationship of vehicle tank vent pressure versus tank weight increase is presented in Figure F-13.



External profile of booster.

FIGURE F-7



Internal profile of booster.

FIGURE F-8

Mass Characteristics		Staging (Booster Burnout)	Initiation of Jet Powered Flight
Weight (lb)		474,876	451,219
Center of Gravity Location (in.)	X	-2010	-2004
	Y	0	0
	Z	13	14
Moment of Inertia (slug — ft^2 $\times 10^6$)	Ixx	7.017	7.016
	Iyy	53.918	51.25
	Izz	57.013	54.354

MASS CHARACTERISTICS OF BOOSTER

Tankage Characteristics	Liquid Hydrogen Tank, Boost	Liquid Oxygen Tank, Boost	Liquid Hydrogen Tank, Jet Fuel
Volume (ft^3)	82,951	29,885	9692
Pressurization Range (psia)	25-27	25-27	30-32
Vent Pressure Range (psia)	28-30	28-30	34-36
Surface Area (ft^2)	12,520	5030	
Weight (lb)			
Tank	39,524	11,842	
Insulation	3030		

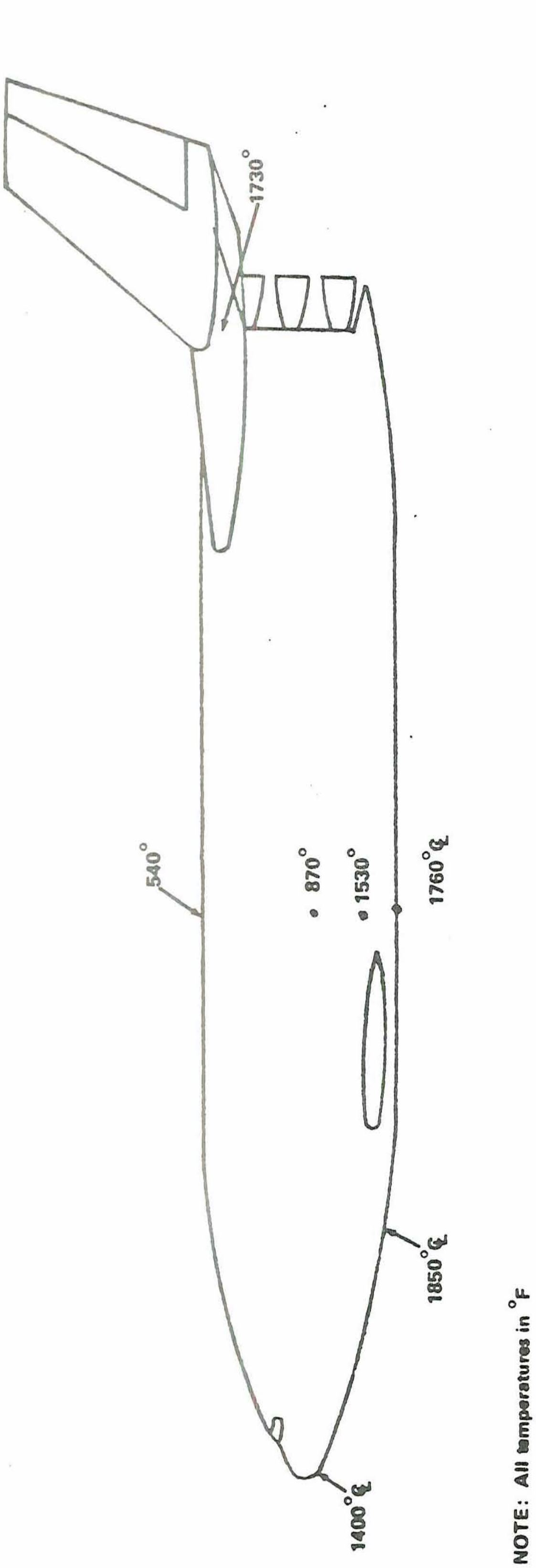
PROPELLANT TANKAGE
CHARACTERISTICS OF BOOSTER

FIGURE F-9

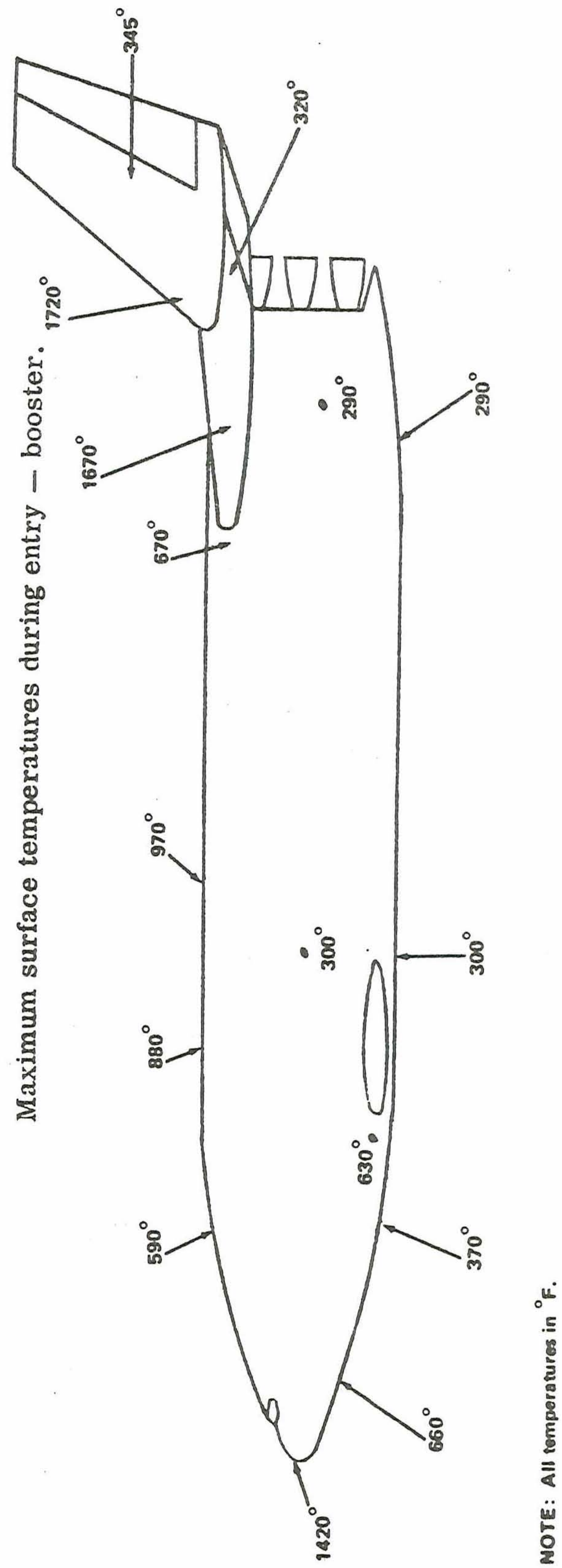
Residual Characteristics	Liquid Hydrogen Boost Tank	Liquid Oxygen Boost Tank
Mass at burnout (lb)	6886	18,796
Mass percent liquid (%)	88	76
Temperature of vapor (° R)	450	520
Total pressure (psia)	26	26
Heat leak into liquid propellant (Btu/hr/ft^2)	100-330	100-1000
Mass of helium pressurant (lb)	0*	0*
*Autogenous pressurization system assumed.		

RESIDUAL PROPELLANTS OF BOOSTER

FIGURE F-10



Maximum surface temperatures during entry — booster.



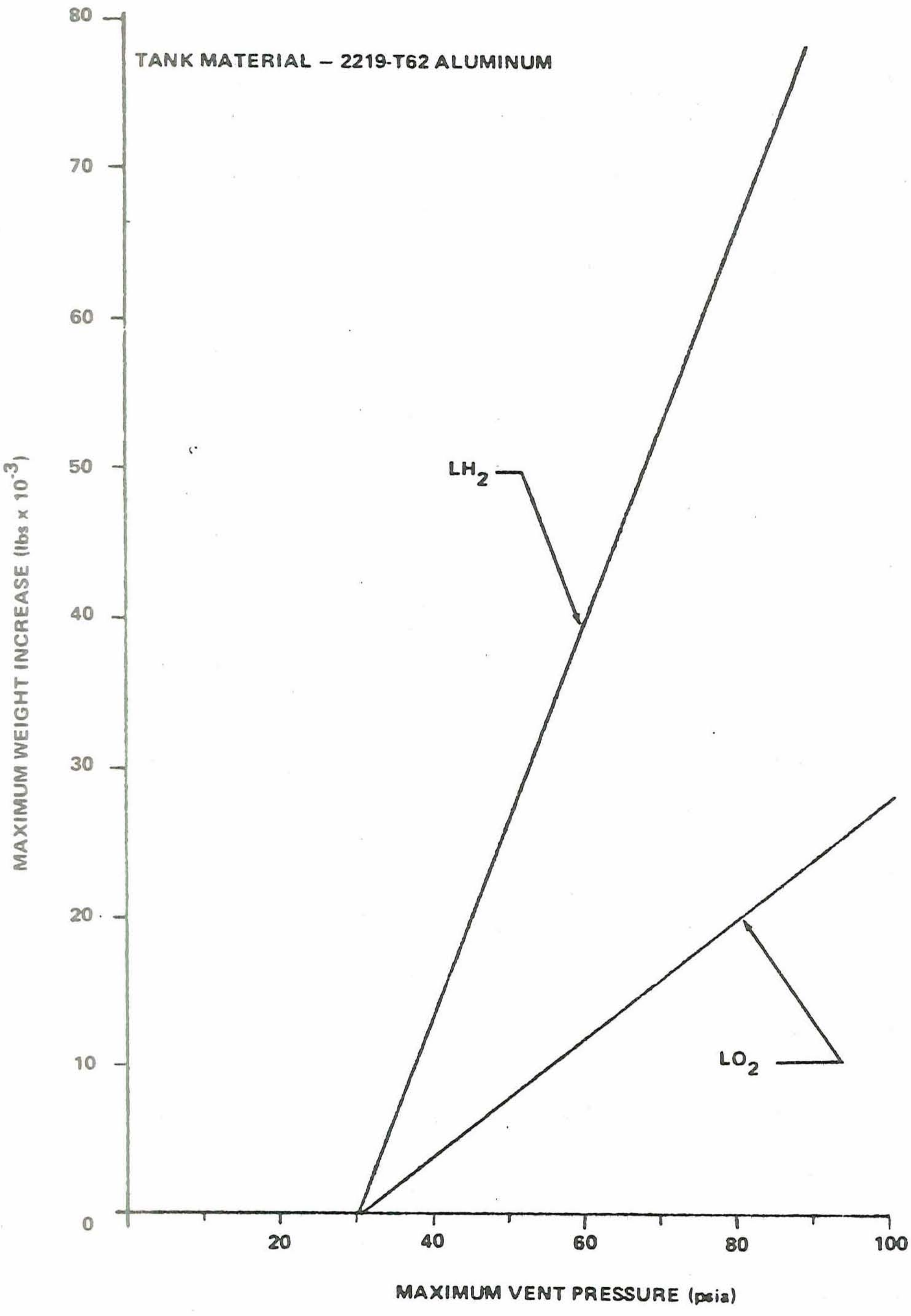
Maximum surface temperatures during ascent — booster.

FIGURE F-11

Thruster	Location (in.)			Thrust Vector Direction Cosine		
	X	Y	Z	X	Y	Z
Pitch	-858	12	193	0	0	-1
Pitch	-858	-12	193	0	0	-1
Yaw	-858	192	12	0	-1	0
Yaw	-858	192	-12	0	-1	0
Pitch	-858	12	-190	0	0	1
Pitch	-858	-12	-190	0	0	1
Yaw	-858	-192	12	0	1	0
Yaw	-858	-192	-12	0	1	0
Roll/Pitch	-2658	552	215	0	0	-1
Roll/Pitch	-2658	552	154	0	0	1
Roll/Pitch	-2658	-552	215	0	0	-1
Roll/Pitch	-2658	-552	154	0	0	1
Yaw	-2778	745	195	0	-1	0
Yaw	-2796	745	195	0	-1	0
Yaw	-2778	-745	195	0	1	0
Yaw	-2796	-745	195	0	1	0

AUXILIARY PROPULSION SYSTEM
THRUSTER LOCATIONS – BOOSTER

FIGURE F-12



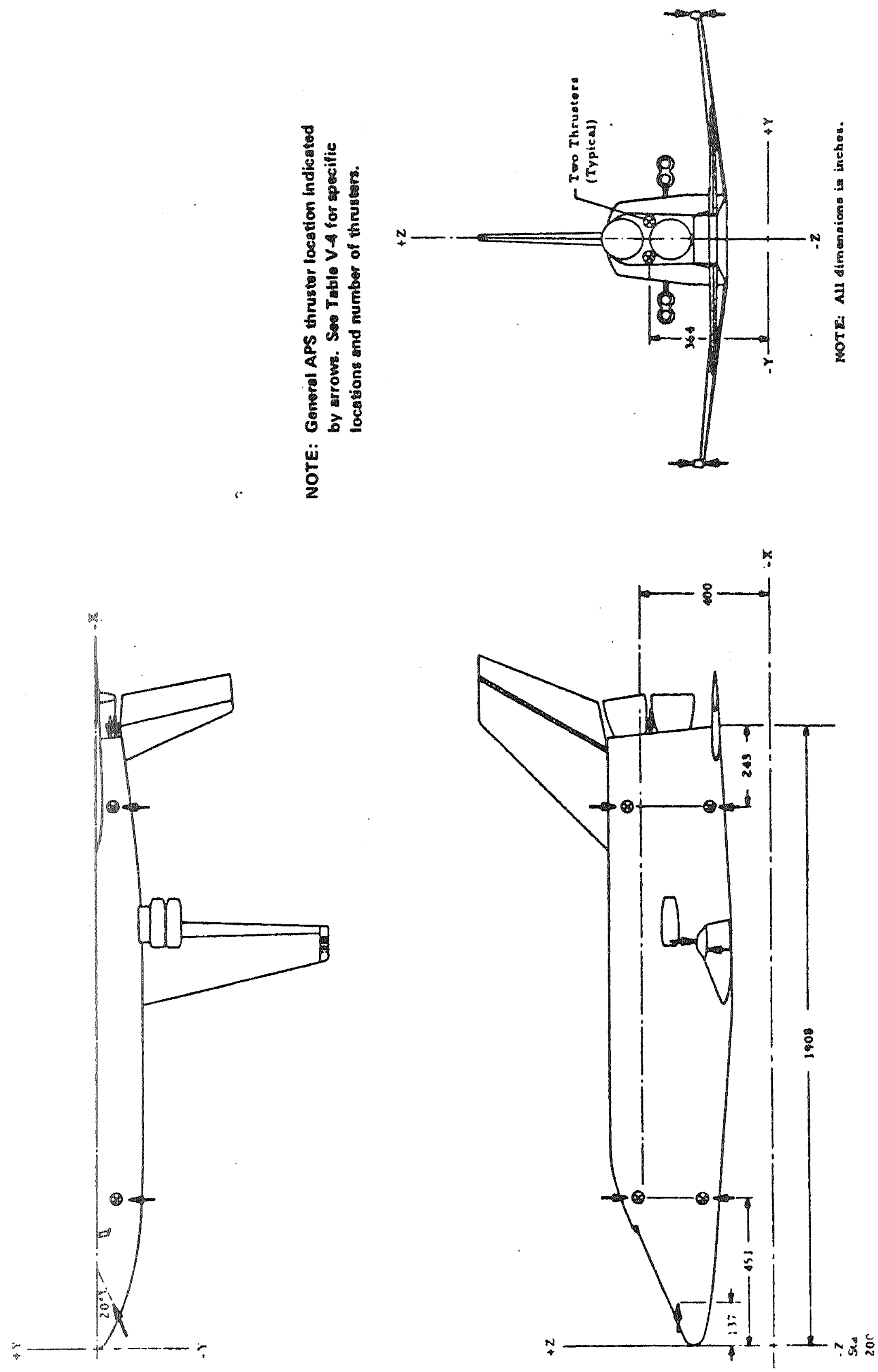
Tank weight sensitivity — booster.

FIGURE F-13

F-5. ORBITER B CHARACTERISTICS

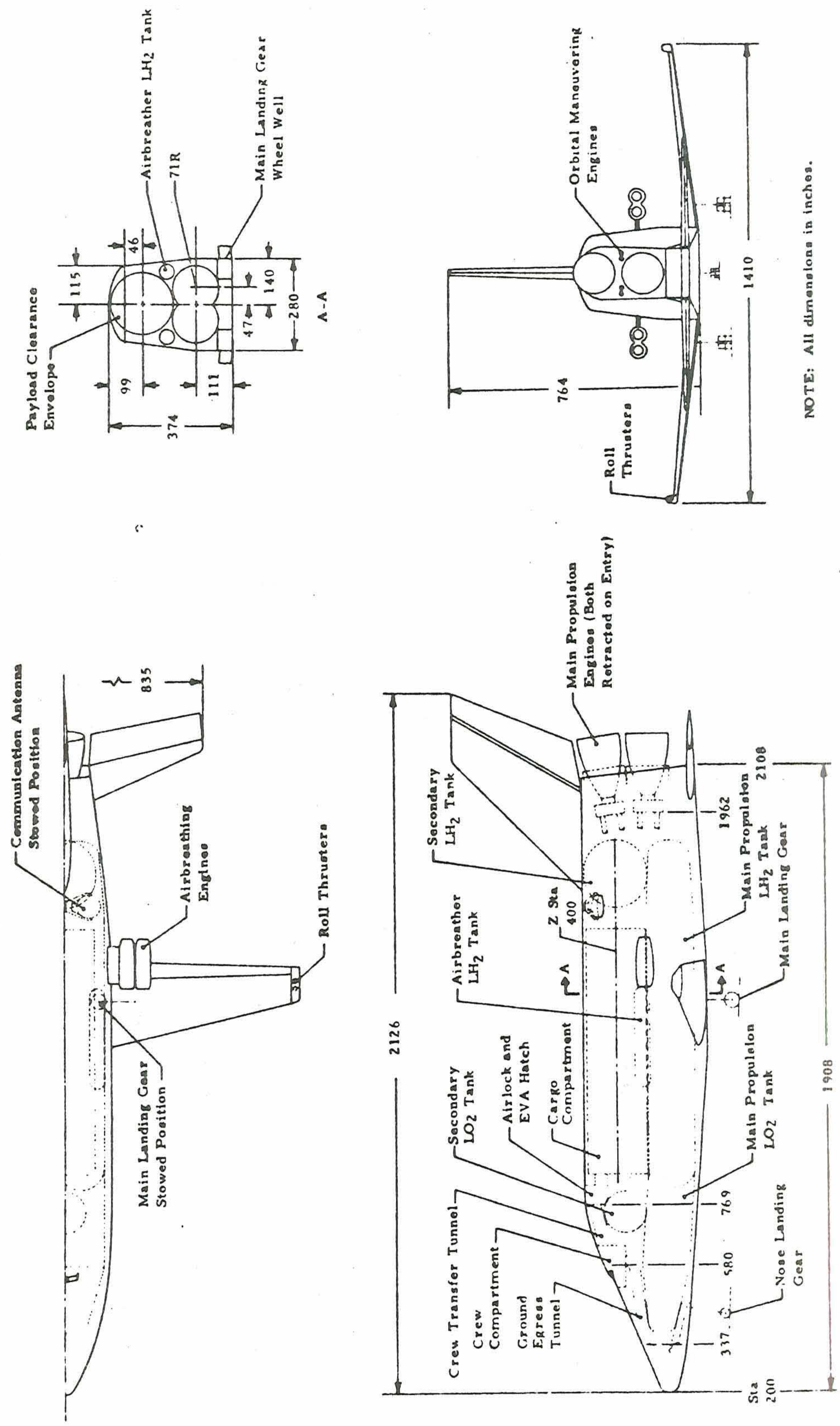
Orbiter B is a straight-wing, low cross-range stage. The orbiter weight at booster/orbiter separation is approximately 665,000 lb. Airbreathing engines are located in pods on each side of the fuselage for flyback and ferry operations.

External and internal profile information is presented in Figures F-14 and F-15. Further information on mass characteristics, tankage, propellant residuals, and heating environments is presented in Figure F-16 through F-18. Auxiliary propulsion thruster locations are listed in Figure F-19.



External profile — orbiter B.

FIGURE F-14



Internal profile — orbiter B.

FIGURE F-15

Description	Nominal Weight (lb)	Center of Gravity (in.)			Moment of Inertia (slug-ft^2 $\times$ 10^6)		
		X	Y	Z	Roll	Pitch	Yaw
Launch	737,211	-945	0	262	2.052	32.123	32.385
50% Burn	504,067	-1082	0	274	2.035	23.708	23.987
Injection	271,465	-1259	0	300	1.950	15.074	15.438
Maneuver	253,613	-1275	0	299	1.933	14.177	14.559
Preretro	241,997	-1293	0	295	1.915	13.384	13.782
Reentry	234,400	-1305	0	293	1.907	12.951	13.358

MASS CHARACTERISTICS OF ORBITER B

Propellant	Weight (lb)	Volume (ft^3)	Surface Area (ft^2)	Vent Pressure (psia)
Liquid Oxygen	3934 (Tank) 1045 (Common Bulkhead)	5957	2048	26-30
Liquid Hydrogen	7268 (Tank) 1735 (Insul.)	16,098	5122	26-30

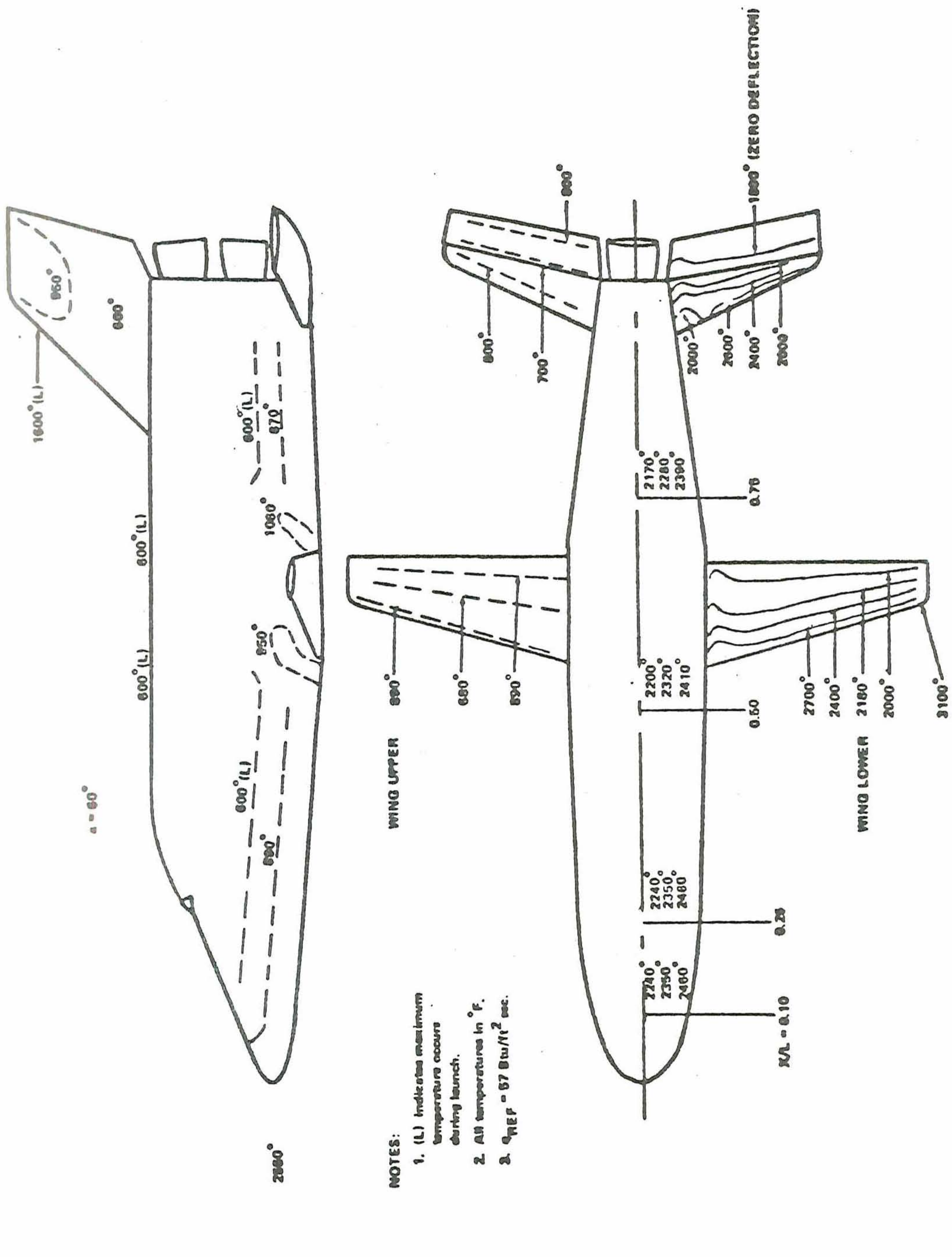
PROPELLANT TANKAGE CHARACTERISTICS OF ORBITER B

FIGURE F-16

Residual Characteristics	Hydrogen Boost Tank	Oxygen Boost Tank
Liquid Mass at burnout (lb)	736	2440
Vapor mass at burnout (lb)	167	1191
Helium mass in the tank (lb)	2.95	1.18
Vapor temperature at burnout (°R)	454	379
Tank pressure at burnout (psia)	26	26
Heat rate into tank skin (Btu/hr/ft^2)	7-36	4-TBD

RESIDUAL PROPELLANTS OF ORBITER B

FIGURE F-17



Maximum outer skin temperatures — orbiter B.

FIGURE F-18

Thruster	Thruster Location (in.)			Thrust Vector Direction Cosine		
	X	Y	Z	X	Y	Z
-X Translation	-337	+55	+300	-0.940	-0.342	0
-X Translation	-337	-55	+300	-0.940	+0.342	0
Yaw	-651	+97	+406	0	-1	0
Yaw	-651	-97	+406	0	+1	0
Yaw	-651	+135	+218	0	-1	0
Yaw	-651	-135	+218	0	+1	0
Pitch	-651	+56	+442	0	0	-1
Pitch	-651	-56	+442	0	0	-1
Pitch	-651	+135	+186	0	-0.707	+0.707
Pitch	-651	-135	+186	0	+0.707	+0.707
Roll	-1429	+693	+228	0	0	-1
Roll	-1429	-693	+204	0	0	+1
Roll	-1450	+693	+204	0	0	+1
Roll	-1450	-693	+228	0	0	-1
Yaw	-1865	+92	+444	0	-1	0
Yaw	-1865	-92	+444	0	+1	0
Yaw	-1865	+112	+192	0	-1	0
Yaw	-1865	-112	+192	0	+1	0
Pitch	-1865	+50	+485	0	0	-1
Pitch	-1865	-50	+485	0	0	-1
Pitch	-1865	+112	+160	0	-0.707	+0.707
Pitch	-1865	-112	+160	0	+0.707	+0.707
+X Translation	-2102	+45	+364	1	0	0
+X Translation	-2102	+63	+364	1	0	0
+X Translation	-2102	-45	+364	1	0	0
+X Translation	-2102	-63	+364	1	0	0

AUXILIARY PROPULSION SUBSYSTEM THRUSTER
LOCATIONS, ORBITER B

FIGURE F-19

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY • EAST
Saint Louis, Missouri 63166   314·232·0232